



Università degli Studi di Padova
Dipartimento di Ingegneria Industriale
Corso di Laurea in Ingegneria Aerospaziale
Tesi di Laurea Magistrale

**"CONCEPTUAL DESIGN OF A FIXED-WING
UAV FOR MARS RECONNAISSANCE"**

Relatore:

Prof. Giacomo Colombatti

Laureando:

Pietro Pinton

Mat. 2130544

Abstract

As the exploration of Mars transitions from probes and rovers to the possibility of human missions, the focus must shift from slow scientific exploration to rapid and efficient reconnaissance. Current orbiters with limited coverage and slow rovers leave a critical gap in situational awareness and exploration capabilities for future astronauts. This thesis proposes the conceptual design and a feasibility analysis of a reusable, rocket-propelled Unmanned Aerial Vehicle (UAV) tailored for the thin atmosphere of Mars.

Moreover, the study integrates aerodynamic sizing with numerical flight dynamics simulations based on real MOLA topographic data. The results validate the effectiveness of the proposed design: simulation data show a liftoff distance of just 217 m and an operational range of 1500 km , while offering an endurance exceeding 4.5 hours.

Ultimately, this work demonstrates that by using unconventional launch strategies and simple but reliable aerodynamic design, this aerial platform can effectively serve as a long-range scout, bridging the gap between orbit and ground to ensure the safety of future astronauts.

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1 | Introduction

The latest Mars exploration mission, *Mars 2020*, featured the first controlled aircraft flight in a non-terrestrial atmosphere. A small rotary-wing drone, named *Ingenuity*, completed 72 flights in the Jezero Crater area, achieving a total flight time of 122.8 minutes, and covering a total of approximately 17 *km*, while reaching a maximum altitude of 24 *m*. This huge success is somewhat comparable to the Wright brothers' first 12-seconds flight with their *Flyer* in 1903, which marked the beginning of over a century of aviation innovation and growth that continues today.

Just as that 1903 flight ignited the growth of the entire aeronautical sector, *Ingenuity* has paved the way for a new era of space exploration, which is no more only defined by probes or small ground rovers, but also by actual aircraft. These vehicles can expand research areas on an unprecedented scale, potentially covering hundreds or even thousands of kilometers, gathering the same amount of data on a single flight that would take a traditional mission years.

This is what sparked the idea for this thesis: to propose, even if at an initial stage, a concept for a high-performance fixed-wing drone; the goal is to shorten exploration timelines on Mars and support a future crewed mission by performing reconnaissance, mapping, and other various tasks.

Mars' environment, however, is very hostile, meaning that the design challenges for such a system are numerous and complex, and this document will only address them partially and at a conceptual stage. In fact, results presented in this research are still far from being valid for a final design; however, the engineering methodology followed was structured to be as rigorous and realistic as possible.

2 | History and Current Status of Martian Exploration

Since ancient times, Mars's reddish color has made it peculiar in the night sky; this led various cultures to associate it with the image of blood and war, but also of fertility and agriculture, such as the Roman god Mars [1].

2.1 The Pre-Space Era

When the telescope was invented and adopted as a scientific instrument, the first direct observations of the surface began. Among the many scientists and astronomers who dedicated themselves to this cause was Giovanni Schiaparelli, in the nineteenth century, despite his discoveries falling victim of a linguistic misunderstanding that fueled the myth of Mars as an inhabited planet. During Schiaparelli's multiple observations of the planet, he discovered a series of thin lines on the planet's surface, which he described using the Italian word "canali". The problem arose when this work was translated to English using the word "canals," which implies an artificial and anthropic origin, rather than the correct word "channels", which would have indicated the more natural origin that Schiaparelli suggested [1].

This seemingly minor misunderstanding gave birth to a long series of speculations about the planet's past and present, and many scientists and authors came to believe it was inhabited by a population extremely similar to humans. It would take the arrival of the space age and the first close-up observations of the planet to finally correct these wrong assumptions.

2.2 The Beginning of the Space Age

The scientific exploration of Mars began in the 1960s. This was the golden period of the space race, set within the broader context of the Cold War, which saw multiple unmanned probes being sent to the Red Planet, both by the USA and the USSR. The first close-up, black-and-white images arrived in 1964 from the American *Mariner 4* probe; for the first time, it was possible to conclude with certainty that Mars was indeed an arid and uninhabitable planet, not the Earth-like one with a humanoid race inhabiting it, as some liked to believed [1]. This mission also collected the first data regarding the planet's atmosphere.

Numerous subsequent missions, however, confirmed the presence of hydro-geological channels on the surface. The question therefore remained: could Mars, in its remote

past, have possessed the conditions to host life forms?

2.3 The *Viking* program

When the *Mariner* program was concluded, the focus shifted to searching for traces of life on the surface, whether present or past. The *Viking* program, launched in 1975, consisted of two identical missions: *Viking 1* and *Viking 2*, both composed of an orbiter and a lander segment. The two landers took high-resolution images of the terrain during their descent, and, once on the ground, they conducted environmental analyses and performed several experiments aimed at detecting traces of microbial metabolism in the soil. Results proved that, if life ever existed on Mars, it had almost certainly been extinct for a long time; this meant the search had now to shift to areas that were more likely to have been habitable in the past [1].

2.4 The Era of Rovers

Following the loss of the *Mars Observer* mission in 1993, the era of Martian exploration as we know it today, began. The *Mars Pathfinder* mission, which landed on the planet in 1997, deployed the first small wheeled rover, *Sojourner*; it only covered about one hundred meters in total during this first, demonstrative mission lasting 83 sols, but since it could move away from the landing site, it performed surveys and terrain analysis at 16 different points near the main lander. The mission also took over 500 photographs, providing important evidence of past liquid water on Mars [1].

The year 2004 was a particularly intense one, as the twin rovers *Spirit* and *Opportunity* landed on the planet just three weeks apart. Both were designed for a brief 90-sol operational life, but they far exceeded expectations and continued their activities for years. Unlike *Sojourner*, which was effectively an appendage of the *Pathfinder* lander, capable of working only in its proximity, *Spirit* and *Opportunity* were completely independent mobile laboratories, with their lander merely serving as a protective shell for atmospheric entry and landing. This allowed the rovers to follow a linear, continuous exploration path without needing to stay near their landing site.

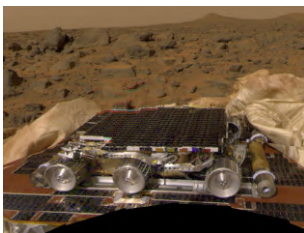
During its incredible 2208-sol operational life, *Spirit* traveled for 7730 m, while facing the many hazards of the Martian environment, including low temperatures, dust settling on its solar panels, and sandy, unstable terrain. Despite this, it made an inestimable contribution to research by finding deposits of silica and carbonates, a strong indication of interaction between hot water and rocks. The mission officially ended in 2011, more than a year after the last successful contact with the rover [31].

However, its sister rover, *Opportunity*, did even better: it continued its mission for 5353 sols, capturing some of the most iconic photographs of Mars and setting the still unbeaten record for distance traveled on another planet of over 45 *km*. Contact with *Opportunity* was unfortunately lost in 2018 due to a massive sandstorm which likely made impossible for the rover to charge its batteries using solar panels [31].

2.5 The State of the Art: *Curiosity* and *Perseverance*

Thanks to the experience gained from previous missions, the *Curiosity* rover was launched in 2011: it is a true mobile laboratory, capable of performing advanced chemical analyses and experiments completely autonomously. The mission is still ongoing despite the several challenges of the hostile Martian environment. To date, it has collected important information on the past of Gale Crater [32].

The most recent mission, which carried the *Perseverance* rover, based on its sibling *Curiosity*'s hardware, landed in 2021. Its objective is to search for biosignatures (i.e. chemical traces of microbial life) in the Jezero Crater; this is a geological site of primary interest because it is believed to have hosted a river delta in the remote past, making it an ideal place for life to have developed. The rover features state of the art scientific instruments, as well as the ability to store samples for future retrieval. The most notable feature of the mission, however, is what *Perseverance* carried under its belly: the drone *Ingenuity*, a small helicopter thanks to which the first controlled flight in another planet's atmosphere was completed in April 2021, adding a new dimension to the exploration of Mars. *Ingenuity* was intended as little more than a technology demonstrator, but it vastly exceeded initial expectations. It completed 72 flights, covering a total of over 17 *km*, and logged a total flight time of over two hours. This experience not only proved the feasibility of flight in the Martian atmosphere but also highlighted the many challenges that must be overcome to benefit from it in the future. The drone unfortunately suffered a broken blade in January 2024, possibly due to a guidance error, and is now used as a stationary weather station, being unable to fly [31].



(a) The *Sojourner* rover



(b) *Ingenuity* and *Perseverance*

2.6 Possible Use of a Fixed-Wing Drone

The experience gained from the *Perseverance* mission naturally leads to questions about the future of Martian exploration: from a Global Positioning and Navigation System, which likely could have prevented the *Ingenuity* accident [31], to a human crewed mission or even a small settlement on the ground or in orbit, the future is yet to be written.

The technological solution presented in this document would require human personnel on the planet to be used correctly; this means it is not a feasible proposal for the short term. However, this is not the first time a fixed-wing aircraft for Mars exploration has been proposed: the *ARES* (Aerial Regional-scale Environmental Survey) mission, developed in the early 2000s, planned to use a rocket-powered aircraft for large-scale geomagnetic surveys and atmospheric sampling [16]. Despite the initial enthusiasm, however, the mission was discarded in favor of the *Phoenix* probe, which landed on Mars in May of 2008.

It is still possible that the first use of fixed-wing technology could happen well before human exploration, by releasing the drone directly into the atmosphere, as the *ARES* mission planned to do [24]; clearly, this would limit the mission to a single flight, as there is currently no way to identify a safe site for autonomous landing, recharging and taking off again, due to the rugged surface of the planet.

Regardless, the enormous potential of a fixed-wing drone must not be ignored. To

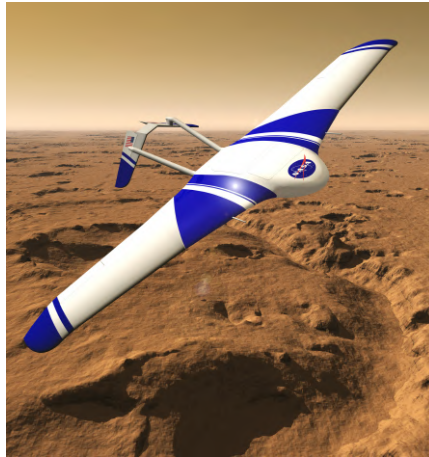


Figure 2.2: The proposed design for *ARES*

date, the total distance covered on Mars by all rovers combined does not exceed the 100 *km* mark [31]; on the other hand, an optimized aircraft could potentially cover multiple times that distance on a single flight before returning to base for future reuse. A mission of this kind would have massive potential for ground mapping, searching for geological features, geomagnetic surveys, atmospheric studies, and countless other applications.

A fixed-wing drone has been selected, since it is more energy efficient than a rotary-wing drone, especially during cruise flight; this is because the propulsion system only needs to generate enough thrust to overcome aerodynamic drag, whereas the wings generate the necessary lift, while rotary-wing system does not have this advantage. Furthermore, fixed-wing systems generally have a greater payload capacity, and their only real disadvantage compared to rotary-wing systems is the inability to land in rough terrain [2].

The drone itself would be a reusable and largely autonomous vehicle. It could assist other autonomous units, such as rovers or other drones through reconnaissance, but it could also provide support to human users, including cargo transport. The aircraft presented in this document has a payload capacity of 20 kg for basic scientific instruments. However, nothing prevents replacing that mass with supplies. Alternatively, an ad hoc cargo version based on the same geometry could be created, trading slightly lower performance for greater payload capacity.

3 | The Martian Environment

Mars features a fascinating but complex topography, shaped over time by slow hydro geological phenomena and impact cratering. We are used to seeing it as a barren desert, yet it reveals a past that was much more Earth-like, featuring watercourses, large lakes, canyons, and plains. These environments might have even been capable of hosting life [1].

While this ancient past is intriguing, the current reality that an UAV must face is far more hazardous. The thin atmosphere and reduced gravity alter flight mechanics and impose strict boundaries on the design. For calculations presented in this document, some physical quantities will be treated as constants, while recognizing a certain degree of approximation. They are listed in the table below:

Quantity	Symbol	Value
Mean planetary radius	r_M	3389.5 km
Mean surface gravity	g_M	3.73 m/s ²
Specific gas constant (atmosphere)	R	188.9 J/(kg·K)
Specific heat ratio (for CO ₂)	γ	1.33
Dynamic viscosity	μ	1.2×10^{-5} Pa·s
Kinematic viscosity (at surface)	ν	8×10^{-4} m ² /s

Table 3.1: Characteristic quantities of the planet and the Martian atmosphere [1, 30]

It is also worth noting the duration of the Martian solar day, commonly referred to as a "Sol"; it lasts 24 hours, 39 minutes, and 35 seconds, making it remarkably similar to Earth's day [1]. This similarity is advantageous for mission logistics, allowing potential future human operations to follow a rhythm almost identical to that of Earth.

3.1 Definition of Altitude

Defining altitude relative to "sea level" is a concept that applies only to Earth, and on a rocky planet like Mars, which has no seas, a specific geodetic zero is required. To address this, a Martian datum was introduced; this zero reference level was originally defined as the average altitude with an atmospheric pressure of 610 Pa [1]. Modern satellite mapping and ground survey missions have allowed for a more precise definition, and the datum is now defined as an equipotential surface that coincides, on average, with the original pressure reference. All altitudes reported in this document will be relative to this datum.

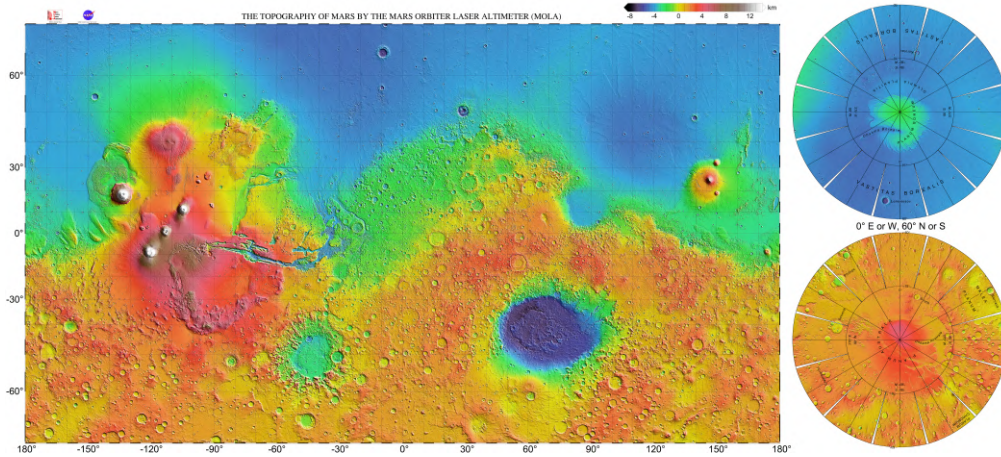


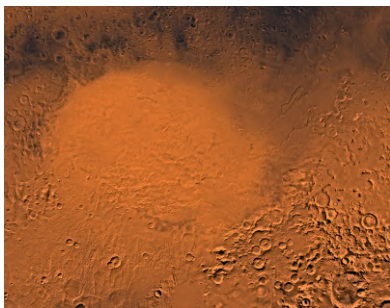
Figure 3.1: Mars Topography

3.2 Martian Geography

Even a quick glance at satellite imagery reveals the sharp contrast between the northern and southern hemispheres of the planet. The North is dominated by vast plains, with altitudes reaching thousands of meters below the datum. The South, in contrast, is visibly more mountainous, rugged, and heavily cratered, with an average elevation up to 8 km higher than the northern hemisphere.

Mars also hosts truly spectacular and unique geological formations: *Olympus Mons* stands 22 km high, making it the tallest mountain in the Solar System; conversely, *Valles Marineris* is a vast canyon system 4000 km long, reaching depths of 7 km . Another notable feature is *Hellas Planitia* in the southern hemisphere: a vast plain located within a huge impact crater that spans over 2300 km .

Both poles feature extensive ice caps. These are composed of water ice, making them a primary resource of interest, and dry ice which sublimates and deposits as frost depending on the season [1].



(a) Hellas Planitia



(b) Olympus Mons

Figure 3.2: Martian landscapes

3.3 The Atmosphere and its Exponential Model

Mars' atmosphere has been the subject of recent studies by the SAM (*Sample Analysis at Mars*) module, installed on the *Curiosity* rover. This instrument measures the concentration of isotopes and volatile compounds, as well as detecting carbon-based compounds, with the ultimate goal of determining if and when the atmosphere was capable of supporting life [32].

Currently, however, the Martian atmosphere is extremely rarefied. It has an average surface pressure of just 610 *Pa* and a reference density of only 0.02 *kg/m*³; this makes the environment completely inhospitable and, crucially for this research, poses significant challenges for atmospheric flight [30]. The main component of this thin atmosphere is carbon dioxide (CO₂), which accounts for 95.1%. Other components include nitrogen (2.59%), argon (1.94%), oxygen (0.161%), and carbon monoxide (0.058%). These percentages can vary slightly depending on the location and the season.

To correctly analyze flight dynamics, it is needed to define an atmospheric model that could accurately describe how variables like density, pressure, and temperature change with altitude. The reference conditions are listed on the table below:

Quantity	Symbol	Value
Density	ρ_0	0.02 kg/m ³
Pressure	P_0	610.5 Pa
Temperature	T_0	210 K

Table 3.2: Reference quantities at 0 m of altitude

Since the UAV proposed in this paper will have a nominal cruise altitude of 1000 *m*, the atmospheric model only needs to be reliable at relatively low altitudes. An extensive analysis of the entire atmosphere is unnecessary for this investigation, thus a simple exponential atmospheric model was created. For the sake of simplicity, it was assumed that the temperature remains constant at the altitudes considered ($T_0 = \bar{T}$); this allowed to define the scale height as $H = \frac{R \cdot \bar{T}}{g_M} = 10700$ *m*.

The resulting model is developed as follows:

$$P(h) = P_0 \cdot e^{-\frac{h}{H}} \quad (3.3.1)$$

$$\rho(h) = \rho_0 \cdot e^{-\frac{h}{H}} \quad (3.3.2)$$

This atmospheric model was implemented in all MATLAB scripts used for the analyses described in this document. Comparing these results with real, experimental data reveals minimal error within the first 2000 *m* of altitude, making the model perfectly acceptable for this application. For validation, data generated by this model

were plotted against data from the NASA Glenn Research Center, which uses an atmospheric model based on measurements from the *Viking 1* mission [30].

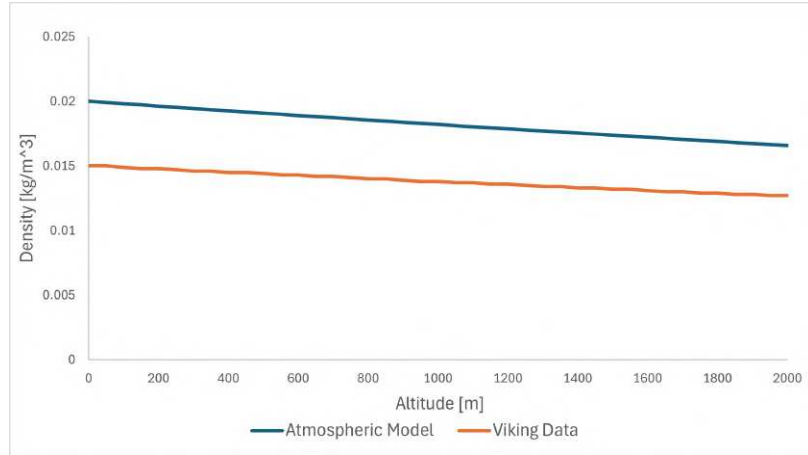


Figure 3.3: Density against altitude

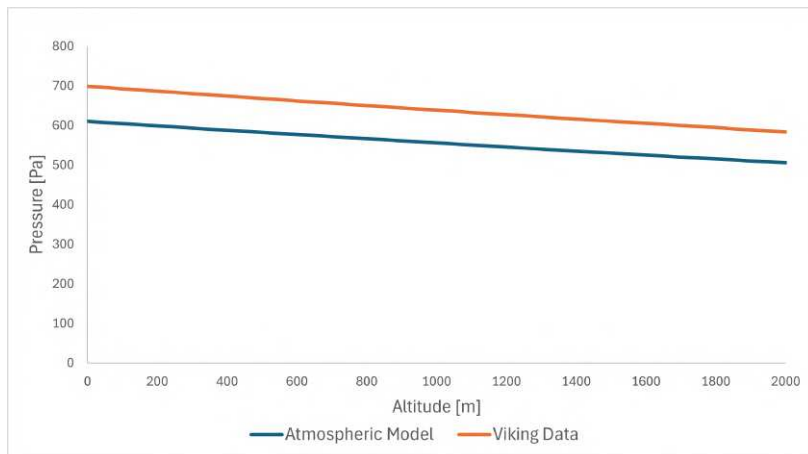


Figure 3.4: Pressure against altitude

Results demonstrate a coherent trend; the slight shift between the two curves is attributed to the difference between the specific, experimental nature of *Viking*'s data and the average reference values used for the model presented here.

3.4 Meteorological Conditions

Martian weather is peculiar and presents unique challenges. These arise from a combination of factors: low atmospheric density, low thermal inertia (which drives massive temperature fluctuations), and dust. Martian wind is driven primarily by thermal gradients and the seasonal sublimation/deposition cycle of CO_2 . Although the atmosphere is very thin, wind speeds can be quite high; however, because the air density is so low, the resulting dynamic pressure is minimal, meaning the wind

poses little structural threat. Despite this, the dust lifted by the wind is a significant factor that must be taken into account: it can obscure solar panels and optical sensors, or infiltrate unprotected mechanical components, potentially compromising their operation.

Throughout the various surface missions, numerous peculiar phenomena have been observed. These range from spectacular *Dust Devils* (small convective vortices capable of lifting dust) to full-scale global sandstorms that can obscure vast areas of the planet for months at a time. Another factor making the Martian surface inhospitable is the lack of atmospheric protection against radiation. The surface is constantly bombarded by ionizing radiation, and the lack of a global magnetic field leaves the planet exposed to cosmic rays.

Finally, due to the greater distance from the Sun, the planet's orbital eccentricity, and the inverse-square law of light propagation, Mars receives an average solar irradiance of only 568 W/m^2 [30]. Compared to Earth's 1361 W/m^2 , this places a significant limit on the effectiveness of solar panels as a primary power source, and it is one of the many reasons why a solar-powered UAV was deemed unsuitable.

3.5 Areas of Interest and Possible Landing Sites

As was already mentioned, the technological solution that is proposed in this document, is not currently feasible as of today without the presence of human astronauts to support and maintain it. The UAV itself is designed to be autonomous in its flight activities, however, it will still need human intervention to be properly refueled and taken care of. Moreover, as it will be further discussed in Chapter 9, the drone cannot operate without a basic landing strip, which will obviously need to be clear of obstacles: as of today, such a vast and intensive operation is difficult to be imagined to be undertaken only by rovers or other autonomous system. Choosing the right operational theater is thus, just as critical as the design of the drone itself, since it will be needed to find an area which combines scientific interest with habitability. Most current interest focuses on the northern hemisphere lowlands: these vast plains generally offer lower elevations and smoother topography compared to the rugged southern hemisphere [1]. For the purposes of this thesis, four specific regions were evaluated based on their geomorphology, potential for In-Situ Resource Utilization (ISRU), environmental conditions and scientific interest.

3.5.1 *Arcadia Planitia*

Situated in the northern hemisphere, *Arcadia Planitia* stands out as one of the most promising locations for future human exploration. It is the primary site selected for

the mission simulation discussed in Chapter 13, and for good reasons: the region is characterized by extensive lava flows that have smoothed out the terrain, resulting in a surface with very few large craters or boulders. Scientifically, this region is very promising, as radar data from the Mars Reconnaissance Orbiter (MRO) has revealed massive deposits of almost pure water ice hidden just a few centimeters beneath the surface dust; this accessibility makes it a prime candidate for ISRU. From an operational standpoint, the latitude (around 40° – 60° N) offers a decent trade-off: it provides sufficient solar irradiance during the summer months, though the winters are harsh, necessitating robust thermal control for any deployed hardware.

Private space company SpaceX has already shown great interest in this region for future *Starship* missions, because of the availability of water, topography and the proximity of interesting features such as *Erebus Montes* [28].

3.5.2 *Acidalia Planitia*

To the east of the Tharsis volcanic region lies *Acidalia Planitia*, instantly recognizable in telescope observations thanks to its distinct dark soil. This region presents a more rugged and geologically complex terrain. It is believed to be the site of ancient oceans or massive flood events, and the landscape is dotted with features that resemble mud volcanoes on Earth. While the potential for finding evidence of ancient bio-signatures here is high, the surface is more rough, making it more challenging to flatten an area wide enough for flight operations. Ice is believed to be found in *Acidalia Planitia* at depths as low as 50 cm, and solar radiation is good, although less abundant due to its average northern latitude.

3.5.3 *Utopia Planitia*

Utopia Planitia is the largest recognizable impact basin on Mars and has a proven track record, having hosted both the *Viking 2* lander in 1976 and the Chinese *Zhurong* rover in 2021. The terrain here is largely flat but textured with depressions, which are clear signs of ground ice sublimating over geological timescales. The availability of water ice here is comparable to *Arcadia Planitia*, confirmed by radar soundings indicating vast underground deposits. Its location in the mid-northern latitudes means it shares similar solar conditions to *Arcadia Planitia*.

Since it has been the scope of two missions already, it is unlikely that it would be selected as a prime objective.

3.5.4 *Vastitas Borealis*

Encompassing the northern polar region, *Vastitas Borealis* is the largest lowland plain on the planet. The primary advantage in this case is altitude: this is the lowest region on Mars ($4 - 5\text{ km}$ on average below the datum), meaning the atmospheric density is at its highest. For a winged aircraft, this extra air density provides a significant margin of safety for lift and stall speeds; however, the environment is extreme. Being adjacent to the North Polar Cap, temperatures are consistently lower than the other three sites, and the availability of solar radiation is strictly seasonal: during the polar winter, the sun does not rise, making solar power useless and thermal survival difficult. While the surface is rich in water ice and scientifically fascinating, the operational window would be strictly limited to the summer season.

Region	Latitude (approx.)	Longitude (approx.)
<i>Arcadia Planitia</i>	47° N	175° E
<i>Acidalia Planitia</i>	50° N	340° E
<i>Utopia Planitia</i>	46° N	119° E
<i>Vastitas Borealis</i>	$> 70^\circ\text{ N}$	Circumpolar

Table 3.3: Approximate coordinates of possible landing sites

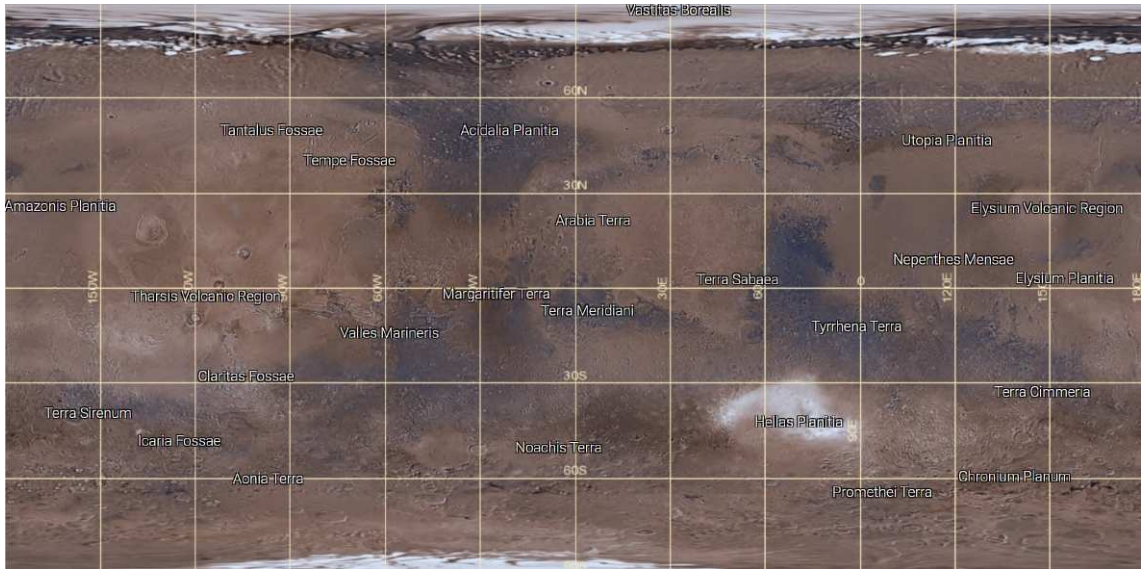


Figure 3.5: Possible landing sites on a topographic view

4 | Effects of Mars Atmosphere on Aerodynamics

Mars atmosphere is characterized by very low density, low pressure and low temperatures, even in full sunlight. It has been already shown how the density, already very low at the datum level, decreases with altitude: for this reason, a nominal cruising altitude of 1000 m has been selected for the drone, making it a good tradeoff between area of observation and detail of observation, all while avoiding low-density portions of the atmosphere [30].

4.1 The Reynolds Number

The Reynolds number is a dimensionless quantity with a central role in fluid mechanics, acting as the primary indicator of the flow regime. Physically, it represents the ratio of inertial forces to viscous forces acting within the fluid, and it is mathematically defined by the equation:

$$Re = \frac{\rho v c}{\mu} \quad (4.1.1)$$

Where:

- ρ is the fluid density (kg/m^3);
- v is the flow velocity relative to the object (m/s);
- c is the mean aerodynamic chord of the wing, (m);
- μ is the dynamic viscosity of the fluid ($Pa \cdot s$).

High Reynolds numbers (typical of terrestrial aviation, where $Re > 10^6$) indicate that inertial forces dominate, leading to boundary layers that readily transition to turbulence and remain attached to the surface. On the other hand, low Reynolds numbers imply that viscous forces play a significant role, and dominate over inertial forces; This regime often delays the transition to turbulence but makes the flow highly susceptible to laminar separation, a phenomenon that can drastically degrade aerodynamic performance [25]. For this reason, the correct choice of airfoils is crucial.

As a comparison, the Reynolds number at the same flight speed can be calculated in Earth and Martian atmosphere for the same hypothetical airfoil of mean aerodynamic chord (M.A.C.) equal to 1 m ; results are reported in table 4.1 [30, 4]: It can

Planet	Density	Airspeed	M.A.C.	Dyn. Viscosity	$Re = \frac{\rho v c}{\mu}$
Mars	$\rho = 0.02 \frac{\text{kg}}{\text{m}^3}$	$v = 50 \frac{\text{m}}{\text{s}}$	$c = 1 \text{ m}$	$\mu = 1.20 \times 10^{-5} \text{ Pa} \cdot \text{s}$	$Re \approx 83333$
Earth	$\rho = 1.225 \frac{\text{kg}}{\text{m}^3}$	$v = 50 \frac{\text{m}}{\text{s}}$	$c = 1 \text{ m}$	$\mu = 1.79 \times 10^{-5} \text{ Pa} \cdot \text{s}$	$Re \approx 3421787$

Table 4.1: Reynolds Number Comparison at 0 m altitude

be noted that the Reynolds number on Mars is roughly 50 times smaller compared to the one calculated for terrestrial condition. Thus, the atmospheric flight of the UAV will be at a low Reynolds regime; an example of similar flight conditions is given by RC plane models here on Earth [25].

Some solutions do exist to increase the Reynolds number, but each one of them presents its own drawbacks. The first possible solution would be to increase the mean aerodynamic chord of the wing; this, however, if the same wing area must be maintained, would decrease the aspect ratio of the wing itself ($AR = b^2/S$), consequently increasing the induced drag component [2]. Another intuitive solution, would be to simply fly faster; this, on the other hand, poses the problems of high-speed flight and the possible insurgence of transonic phenomena, which will be discussed in the following section [11].

4.2 Compressibility and Mach Number

The Mach number M is defined as the ratio of the velocity in the fluid to the local speed of sound a , thus $M = v/a$. The speed of sound in an ideal gas is defined as:

$$a = \sqrt{\gamma R T} \quad (4.2.1)$$

Using the Martian constants defined in Chapter 3 ($\gamma = 1.33$, $R = 188.9 \text{ J/kgK}$) and assuming a standard operating temperature of 210 K , we obtain:

$$a_{Mars} \approx 230 \text{ m/s} \quad (4.2.2)$$

This value is significantly lower than the standard speed of sound at sea level on Earth, which is $\approx 340 \text{ m/s}$; this factor must be taken into great consideration, since it presents a serious design challenge. To generate sufficient lift in the thin Martian atmosphere the UAV must fly at a relatively high airspeed, which, as will be shown in following chapters, can reach up to 120 m/s in certain configurations. While this speed is still well below the sonic barrier, the resulting Mach number is

non-negligible, and cannot be ignored:

$$M \approx 0.52 \tag{4.2.3}$$

At $M > 0.3$, the assumption of incompressible flow begins to waver. While the wing itself may operate in a safe subsonic regime, local flow acceleration over the suction side or on local areas of the external surface could reach much higher speeds, and it must be avoided as much as possible. For this reason, as stated in the previous subsection, increasing the speed is not a viable option for increasing the Reynolds number.

This issue poses a threat not only related to the aerodynamic surfaces, but it could be a dominant factor for the propulsion system, especially in propeller-driven configurations. Propeller tips rotate at speeds much higher than the aircraft's forward velocity, and it would be remarkably easy for them to reach transonic or supersonic speeds ($M > 0.8 - 1.0$), generating shockwaves that drastically increase drag and noise while lowering efficiency [23].

As a conservative measure, a speed limit of 140 m/s has been selected for the design proposed in this document, resulting in $M < 0.6$; however, it will be shown that cruising speeds will be much lower than that value, almost never exceeding 100 m/s .

4.3 Consequences on the Airfoil Design

Airfoil selection is a key aspect in the design of any aircraft, since it will have a major impact on all flight characteristics, from the stall angle to the overall efficiency [2]. For this reason, this process was based on existing studies and reports, which investigated deeply in the subject of suitable airfoils for a fixed-wing aircraft in the Martian atmosphere. Luckily, this is an interesting and studied topic, so plenty of documentation is available; in particular, results from experiments conducted in the Mars Wind Tunnel at Tohoku University and an analytical study from Prudente and Cavalieri were used [15, 20].

Both these studies agree that the most suitable profile for horizontal and vertical tail surfaces is the NACA0006; in particular, it was proven that an elevator with a thin profile, or even an almost flat plate, has beneficial effects on the pitch control, as shown in Figure 4.1. For what concerns the main wing airfoil, almost all studies focus on high-speed applications, pushing the transonic limit; moreover, it has been proven that cambered airfoils perform better than symmetrical ones, and that sharp leading edges produce high adverse pressure gradients, enforcing flow separation and promoting the shedding of vortices [15]. For these reasons, the most proposed airfoil is a supercritical one, the RAE2822, characterized by a maximum thickness

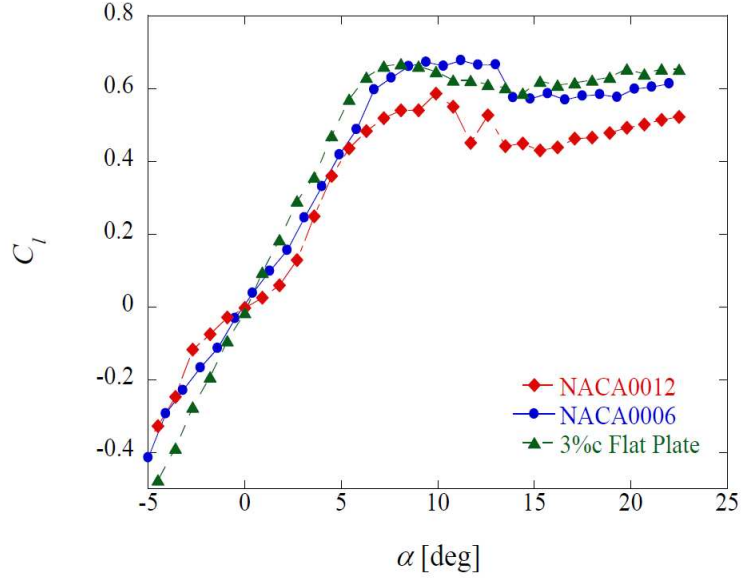


Figure 4.1: Cl - α comparison for possible HT airfoils [15]

of $12.1\% \bar{c}$ and a maximum camber of $1.3\% \bar{c}$ [11]. However, as will be shown in following chapters, the average cruising speed of the proposed UAV design is around $M = 0.35$, so a supercritical airfoil is not really needed, instead it was chosen to adopt the NACA2412, as it has similar geometrical characteristics but, most importantly, it is significantly easier to model and analyze with the limited tools available.

4.4 CFD Simulations

While the choice of NACA0006 for tail surfaces is properly backed up by several studies and scientific reports [20, 15], no real documentation validates the choice of NACA2412 airfoil for the main wing. For this reason, a basic 2D CFD analysis was set up using Ansys Fluent, in which the airfoil behavior in Martian atmosphere at cruise conditions were analyzed: results were encouraging.

In Figure 4.2 it can be seen that, even at a relatively high speed, at around $M = 0.4$ and zero angle of incidence, no shockwave or transonic phenomena are visible; this conclusion was confirmed by looking also at the total pressure contour. Stall behavior was also checked, and results were once again really promising. No flow separation at all was detected for angles of incidence below 8° , and no significant flow separation is visible until $10^\circ/12^\circ$ of incidence; moreover, it must be remembered that in a finite 3D wing, the effective angle of incidence is always smaller, thus making the whole wing more resistant to stall, and its overall behavior more close to experimental results reported in Appendix A [4].

It should be noted that these simulation were performed on a basic level, in order to back up the selection of the main wing airfoil. Despite being executed with an accurate physics model and a well-built mesh, they are just 2D simulations; no 3D CFD simulations were carried out during this conceptual design phase, so the interaction between various parts of the aircraft and their overall behavior was not fully analyzed from this point of view using Ansys Fluent.

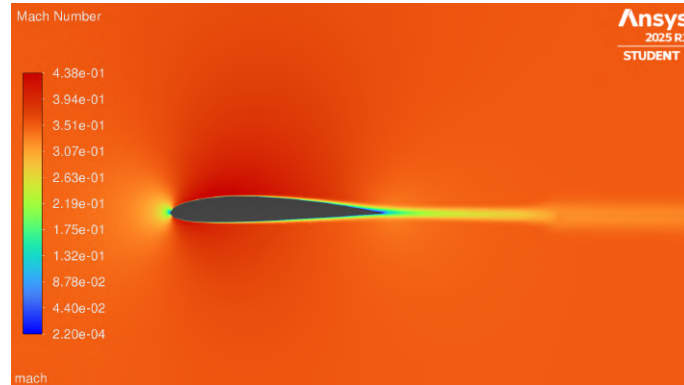


Figure 4.2: Mach number contour for NACA2412 airfoil at 0° incidence

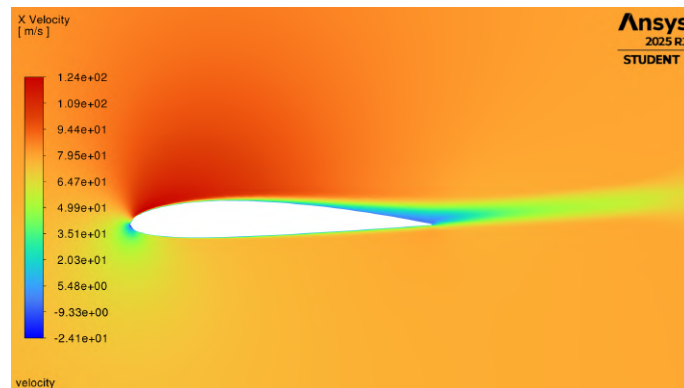


Figure 4.3: Velocity contour at 8° incidence

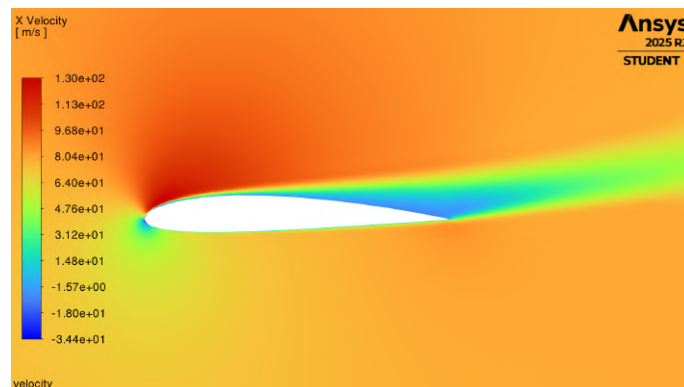


Figure 4.4: Velocity contour at 12° incidence

5 | Systems Engineering Approach

An aircraft is composed of a large number of systems and subsystems, making its design, production, and integration a highly complex undertaking. These challenges become even more apparent when, as in the case of this project, the proposed design is intended to operate on another planet, pushing the limits of state-of-the-art technologies. For this reason, it is necessary to define a series of procedures aimed at organizing the engineering workflow for a project of this magnitude, one that, should it ever become reality, would likely require more than a decade to develop [2]. The methodology adopted for this thesis is based on a realistic Systems Engineering approach. It covers, with some necessary simplifications, the standard activities expected during a Conceptual Design phase [6].

5.1 System Development

In general, the lifecycle of an airplane, or of any engineering project, is structured as follows:

- **Concept:** in this phase, a preliminary concept study is developed based on a set of requirements, with the aim of producing a conceptual design.
- **Definition:** the conceptual input is better defined, and a feasibility study is produced.
- **Design:** in this phase, a detailed design is developed.
- **Production:** the project is physically constructed, based on the inputs produced during the previous phase.
- **Test:** this phase sees the verification of increasingly complex subsystems, up to a test of the whole aircraft. The phase concludes with a certification.
- **Operate:** the design is ready to be operated as intended.
- **Disposal or Retirement:** the end of the operational life of the product.

During all of these phases, one of the most important aspects is the production and preservation of documentation. Particularly in long and innovative projects that could span for years, it is essential to be able to track every single step back to its origin, for tracking development as well as defining responsibilities. Poor documentation can hide some important issues in the project.

Moreover, while the scheme above is not to be followed strictly, as some phases

may overlap each other, it is suggested to. In fact, a proper order and method in project, is useful to keep track of what is being done, what is left to do and, most importantly, it helps to spot issues as soon as possible. Correcting an error can be a minor inconvenience in the Definition Phase, but could turn into an expensive undertaking when production is already under way.[2].

5.2 The Concept Phase

During this phase, a lot of inputs from many different sources come together in order to develop a concept study; usually the this preliminary study is developed around the customer requirements, but it could also be generated from a market study or, as in the case of this thesis, from a simple list of generic specifications.

In this phase, it is necessary to consider many different aspects, thus a Concept Team is usually gathered to handle this amount of data, and to generate a large number of ideas which are collected in the Ideas Pool. The process is cyclical, and it can originally produce an array of possible design solutions; these are then carefully evaluated, and the best one is selected as the design to follow up on. A crucial aspect is not to discard anything that was not selected, since it can be useful in the future for a similar project, or as a baseline for an innovative idea. It must be remembered that it is not certain that a conceptual idea will become a final product, but it surely is a starting point.

To support and validate its work, the Concept Team usually generates reports, drawings, mathematical models as well as commercial material; this is exactly what has been done in this thesis.

5.3 Requirements

Requirements provide a quantitative definition of what the system must do. If clearly defined, they serve as the benchmark to evaluate and determine if the system is compliant with expectations. Along with design drivers, they constitute the foundation of all engineering work required to accomplish the project.

Defining requirements is a crucial process. They must be traceable across various levels; moreover, a good definition allows for the clear identification of the project's critical points and highlights which areas require the most focus. In general, every project aims to achieve a specific objective, and the final product needs the necessary functionalities to do so; thus, a set of requirements serves to indicate what the system must do.

According to standard Systems Engineering practice, a well-written requirement should be: justifiable, unique, unambiguous, traceable, and verifiable. Require-

ments can be classified in various ways; however, one of the most common is the following:

- **Functional Requirements:** These define the desired functionalities or actions to be performed in order to achieve the system's objectives.
- **Performance Requirements:** These quantify the functional requirements, setting specific metrics and levels to be achieved.
- **Design Requirements:** These define the design aspects that must be satisfied; they are often related to boundary conditions such as the flight environment, standard interfaces, or the interaction between subsystems.
- **Operational Requirements:** These specify the conditions required for the system to operate safely, reliably, and within its intended logistics chain.

For the project discussed in this document, the following requirements list was established:

- **F.1:** The system shall provide high-resolution optical and multispectral imagery of the Martian surface.
- **F.2:** The system shall be capable of fully autonomous flight.
- **P.1:** The aircraft shall have a range of at least 1000 km on a single flight.
- **P.2:** The aircraft shall carry a mission payload of at least 20 kg.
- **P.3:** The nominal cruise altitude shall be 1000 m above the ground.
- **D.1:** The system shall ensure payload operability within the vehicle's thermal operating envelope.
- **D.2:** The system shall ensure payload operability within the vehicle's internal pressure environment.
- **D.3:** The payload bay must be physically shielded from dust and debris.
- **O.1:** The total mass of the aircraft at launch (MTOW) shall not exceed 300 kg.
- **O.2:** The aircraft shall be fully reusable.

It is important to note that while this list is merely an illustrative example, and nowhere near exhaustive enough to cover a real-world project of this magnitude, it serves as a valid representation of how the requirements definition process is structured, and how much it will impact following phases.

5.4 Software Tools

To move from requirements to a physical design, different software tools are used; they can range from basic programs in the early stages of the project, to complex, expensive and high-fidelity tools as the design gets refined. In fact, selecting the right tool for the right phase is crucial: using a high-fidelity CFD solver for initial sizing is inefficient, while using a simple analytical method for final validation is too risky. For this thesis, the following software tools were used, each with its specific set of validity and limitations.

5.4.1 MATLAB

MATLAB served as the backbone for the iterative design process and data post-processing. Custom scripts were written to implement the analytical equations for flight mechanics (such as the Breguet range equation, turn radius, and climb rate) and the Mars atmospheric model described in Chapter 3 [34].

Limitations: The results obtained are strictly analytical. They do not simulate fluid physics but rather calculate performance based on fixed aerodynamic coefficients (C_L , C_D) provided as inputs. Therefore, the accuracy of MATLAB outputs is entirely dependent on the reliability of the input data.

5.4.2 XFLR5

XFLR5 is the primary tool that was used for the aerodynamic design of the lifting surfaces, wing sizing, and stability analysis. It is an analysis tool for airfoils, wings, and planes operating at low Reynolds numbers, which integrates the 2D solver *XFOIL* with 3D Linear Lifting Line Theory and Vortex Lattice Method [35]. While XFLR5 is an industry standard for model aircraft and light UAVs, its application in the Martian environment presents specific challenges that must be considered.

Strengths: The software is exceptionally fast, allowing for the rapid analysis of hundreds of wing configurations. It provides excellent estimates for the lift curve slope ($C_{L\alpha}$), induced drag, and longitudinal stability derivatives, which are essential for the conceptual design phase.

Limitations: Despite its utility, XFLR5 has distinct limitations that affect its accuracy for this specific mission profile:

- **Convergence at Low Reynolds:** The internal 2D solver (*XFOIL*) relies on a viscous-inviscid interaction method. At the low Reynolds numbers typical of Mars ($Re \approx 80,000$), the boundary layer is prone to forming laminar separation bubbles. When these bubbles become unstable or large, the numerical solver often fails to converge, potentially leading to gaps in the polar data.

- **Decoupled Viscous/Inviscid 3D Solution:** XFLR5 calculates 3D wing performance by assuming inviscid flow for the 3D geometry and then superimposing the viscous drag derived from the 2D airfoil sections. This means it cannot simulate complex 3D flow separation, or the interaction between the fuselage and the wing root.
- **Compressibility:** XFLR5 includes a Prandtl-Glauert correction for compressibility, but it cannot model shockwaves or non-linear transonic phenomena [35].
- **Fuselage and Bluff Body Modeling:** Algorithms implemented in XFLR5 are mathematically optimized for thin lifting surfaces, thus, they perform poorly when analyzing blunt bodies like fuselages or external pods. Consequently, the fuselage was excluded from the XFLR5 aerodynamic analysis. This simplification is considered acceptable for this conceptual phase; previous third-parties studies have demonstrated that, generally, the fuselage’s contribution to the total drag budget is secondary and does not fundamentally alter the flight mechanics. However, a fixed contribution was added to the total drag calculation; this value depends on velocity and on a fixed drag coefficient for the fuselage and tail booms, but it does not take into account any interference phenomena. A 3D CFD analysis could generate more comprehensive results.

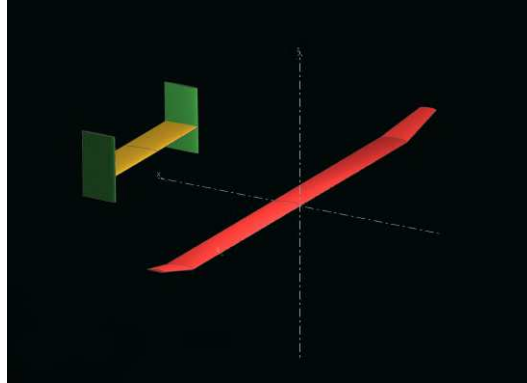


Figure 5.1: 3D visualization of the Twin-Boom configuration in XFLR5

5.4.3 Ansys Fluent

To mitigate the risks associated with the limitations of XFLR5, Ansys Fluent was used. Fluent is a high-fidelity CFD solver capable of solving the full Reynolds-Averaged Navier-Stokes (RANS) equations. It was employed specifically to validate the 2D airfoil performance at the correct Mach and Reynolds numbers, ensuring

that the compressibility effects ignored by XFLR5 were indeed negligible at cruise conditions, as demonstrated in Chapter 4.

Limitations: Due to the high computational cost and the complexity of developing an accurate 3D model of the UAV, the use of Fluent was limited to 2D airfoil validation and was not extended to 3D simulations of the whole UAV.

Some other more basic software tools were used, which were not included in this list as they do not pose a significant threat to the accuracy of the data hereby reported, such as Excel or Prism.

6 | Configuration Selection

Defining the geometry of an aircraft can be quite challenging, especially if a lot of requirements must meet and be satisfied at the same time. Sometimes, it could be necessary to iterate the same design multiple times, finely adjusting it until it performs as intended; this is the workflow that was followed during the design process of this UAV.

6.1 The *Solar Impulse 2* as Baseline

As was mentioned in Chapter 2, some conceptual designs do exist for a fixed-wing Martian drone, however, since none of this was actually fully developed nor proven, an actual baseline does not exist. As a starting point, the experimental plane *Solar Impulse 2* was selected. The first step in the design processes involved a geometric scaling of that architecture: being the first solar-powered airplane to circumnavigate the globe, it weighs more than 2 metric tonnes and its wingspan measures over 77 m, making it incompatible with the requirements of this project. However, its lightness, high efficiency and its low Reynolds number flight regime, made it a good reference architecture [10]. Therefore, the geometry was scaled down to match a theoretical Maximum Take-Off Weight (MTOW) of 200 kg, thus below the maximum allowed TOW, while maintaining the original proportions of the aircraft. Preliminary analysis of this first configuration revealed significant performance issues, particularly tied to such a light mass: to be sufficiently strong, the aircraft needed to be so heavy that no sufficient mass was allocated to fuel, so the design underperformed.

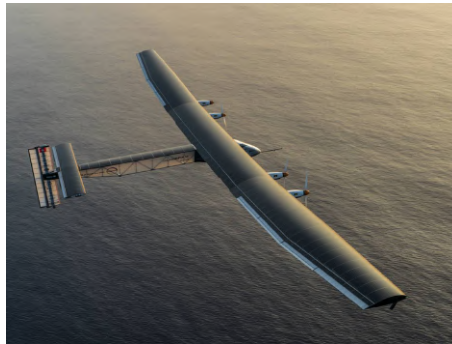


Figure 6.1: The *Solar Impulse 2* HALE aircraft

For this reason, a second version was developed, pushing the limit of the MTOW, bringing the total mass up to 300 kg, while keeping the same geometrical dimensions. This allowed for a stronger structure and greater fuel mass allowed on board, but results obtained in XFLR5 were not positive: the configuration proved rather

unstable in rolling, and it particularly suffered from adverse yaw. Moreover, it did not provide sufficient damping to lateral gusts, which could have resulted in a catastrophic loss of control.

6.2 The Twin-Boom Configuration

To overcome the instability issues of the first designs, two possible design changes were possible: either increase the tail momentum arm, thus resulting in a longer plane, or increasing the rudder surface area [6]. The first option was deemed unsuitable, as an increase in the plane length would have resulted in a higher dry weight, reducing the amount of fuel. Instead, an H-tail configuration was chosen, to allow for a sufficient tail volume to be installed. The resulting geometry was designed in XFLR5 for coefficients and stability analysis; all results were satisfactory, so this was chosen as the final design. Moreover, a rough 3D model was produced using OpenVSP, to allow for a full visualization of the geometry of the UAV [36].

The detailed dimensional parameters are summarized in Table 6.1.

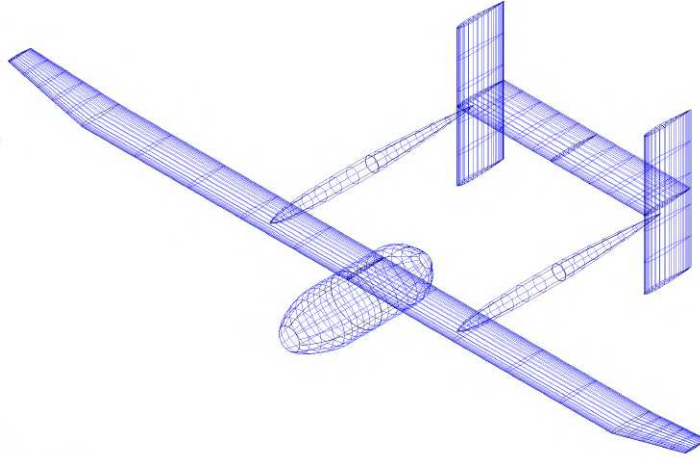


Figure 6.2: The final design

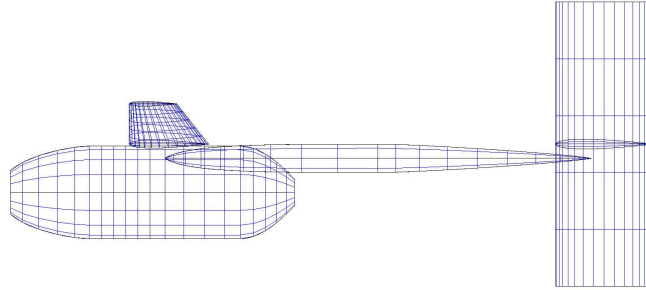


Figure 6.3: Side view of the final design

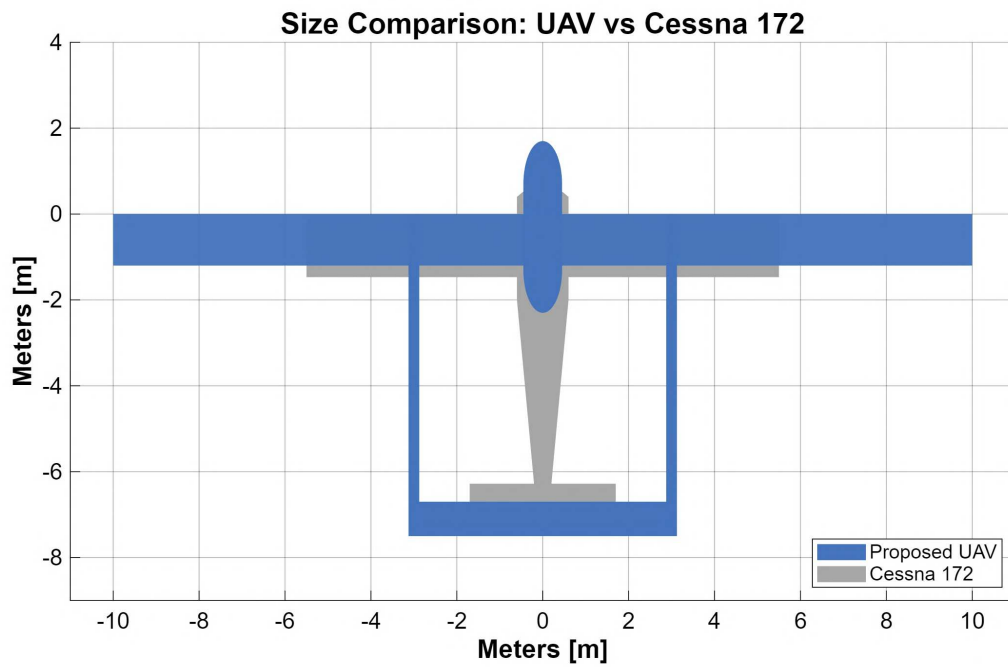


Figure 6.4: Approximate size comparison between the UAV and a Cessna 172

Parameter	Symbol	Value
<i>General Parameters</i>		
Maximum Take-Off Weight	$MTOW$	300.0 kg
Total Aircraft Length	L	9.20 m
Payload Mass	$M_{payload}$	20.0 kg
Fuel Mass	M_{fuel}	120.0 kg
<i>Main Wing Geometry</i>		
Wingspan	b	20.00 m
Wing Area	S	21.34 m ²
Aspect Ratio	AR	18.75
Dihedral Angle	Γ	14.8°
Root Chord	c_{root}	1.12 m
Tip Chord	c_{tip}	0.67 m
Taper Start Position (from root)	y_{taper}	7.78 m
Mean Aerodynamic Chord	$\bar{c}$	1.07 m
Wing Loading	W/S	14.12 kg/m ²
Airfoil Section	-	NACA 2412
<i>Control Surfaces</i>		
Flap Type	-	Plain Flap
Flap Hinge Line	-	80% c
Elevator/Rudder Type	-	Plain
Elevator/Rudder Hinge Line	-	70% c
<i>H-Tail</i>		
Horizontal Tail Area	S_h	8.23 m ²
Horizontal Tail Volume	V_h	2.18
Vertical Tail Area (Total)	S_v	6.86 m ²
Empennage Airfoil	-	NACA 0006

Table 6.1: Geometric specifications of the Twin-Boom configuration

7 | Vehicle Systems and Subsystems

7.1 Material Selection

The primary structural challenge for this Twin-Boom, high-aspect-ratio configuration is not merely ultimate strength, but stiffness; in fact, long, slender wings could be subjected to aeroelastic phenomena such as flutter and divergence. Furthermore, the twin tail booms must not flex excessively, in order to impose the right control authority on the tail.

To meet these requirements, Carbon Fiber Reinforced Polymer (CFRP) was selected as the primary construction material, thanks to its high strength-to-weight ratio. Additionally, the anisotropic nature of carbon fibers allows for aeroelastic tailoring: by optimizing the fiber orientation of the composite layers, it is possible to passively control the twisting of the wing as it bends.

7.1.1 Material Properties

For the preliminary sizing presented in this thesis, a conservative reference density of $\rho = 1600 \text{ kg/m}^3$ for CFRP was selected. The key mechanical properties assumed for the design are summarized in Table 7.1.

Property	Symbol	Range / Value
Density	ρ_{CFRP}	1600 kg/m ³
Tensile Modulus	E_1	100 – 350 GPa
Tensile Strength	$\sigma_{u,1}$	1000 – 3500 MPa
Compressive Strength	$\sigma_{c,1}$	800 – 2000 MPa
Elongation at Break	$\epsilon_{u,1}$	0.8 – 2.0%

Table 7.1: Average Properties of CFRP

Compatibility with the Martian Environment

Beyond providing mechanical performance, the material must survive the hostility of the Martian environment. The choice of CFRP offers excellent compatibility in almost all regards, such as:

- **Thermal Resilience:** Mars experiences extreme temperature swings ($-140^{\circ}C$ to $+20^{\circ}C$). CFRP matrices maintain structural integrity well below cryogenic temperatures, unlike some metallic alloys. Furthermore, the near-zero thermal expansion coefficient prevents thermal stress buildup in the long wing spans.
- **Chemical Inertness:** The CO_2 atmosphere at low pressure ($600 Pa$) poses no corrosion threat. Unlike some metals, CFRP is chemically inert to the Martian environment.
- **Radiation and UV Protection:** The lack of a magnetic field exposes the airframe to high UV levels; while carbon fibers are stable, the polymeric matrix can degrade. To mitigate this, the design must include a UV-resistant coating to shield the resin.
- **Dust and Abrasion:** Martian regolith is abrasive; thus, a sacrificial layer or hardened coating should be applied in order to prevent erosion.

7.2 Dry Mass Breakdown

The mass breakdown of the main structural components was estimated using the semi-empirical method proposed by M.H.Sadraey in his text *Aircraft Design: a Systems Engineering Approach* [2]. This method utilizes semi-empirical equations to estimate component weights based on geometric parameters (such as aspect ratio, wetted area, and taper ratio), the ultimate design load factor (n_{ult}) and the construction material used.

The equations are expressed as follows:

$$\left\{ \begin{array}{l} M_{wing} = \rho_{mat} K_{\rho} S \bar{c} \left(\frac{t}{c} \right)_{max} \left(\frac{AR n_{ult}}{\cos(\Lambda_{c/4})} \right)^{0.6} \lambda^{0.04} \\ M_{ht} = \rho_{mat} K_{\rho} S_{ht} \bar{c}_{ht} \left(\frac{t}{c} \right)_{max} \left(\frac{AR_{ht} n_{ult}}{\cos(\Lambda_{c/4}^{ht})} \right)^{0.6} \lambda_{ht}^{0.04} V_{ht}^{0.3} \left(\frac{c_{elev}}{\bar{c}_{ht}} \right)^{0.4} \\ M_{vt} = \rho_{mat} K_{\rho} S_{vt} \bar{c}_{vt} \left(\frac{t}{c} \right)_{max} \left(\frac{AR_{vt} n_{ult}}{\cos(\Lambda_{c/4}^{vt})} \right)^{0.6} \lambda_{vt}^{0.04} V_{vt}^{0.2} \left(\frac{c_{rud}}{\bar{c}_{vt}} \right)^{0.4} \\ M_{fus} = \rho_{mat} K_{\rho} n_{ult}^{0.25} L_{fus} D_{mean} \end{array} \right. \quad (7.2.1)$$

In which the parameters used are reported in Table 7.2.

Symbol	Parameter Description	Value
<i>Geometric Parameters</i>		
S	Wing Reference Area	21.34 m ²
$\bar{c}$	Mean Aerodynamic Chord	1.07 m
AR	Aspect Ratio	18.17
$(t/c)_{max}$	Max Thickness-to-Chord Ratio	0.12
$\Lambda_{c/4}$	Quarter-Chord Sweep Angle	0°
λ	Taper Ratio	0.6
<i>Physical And Operational Parameters</i>		
ρ_{mat}	Material Density (Carbon Fiber)	1600 kg/m ³
n_{ult}	Ultimate Load Factor	3
<i>Empirical Factors (K_ρ)</i>		
$K_{\rho,wing}$	Wing Density Factor	0.0012
$K_{\rho,ht}$	Horizontal Tail Density Factor	0.032
$K_{\rho,vt}$	Vertical Tail Density Factor	0.09
$K_{\rho,fus}$	Fuselage Density Factor	0.0023

Table 7.2: Parameters used for Dry Mass Estimation

Component Group	Estimated Mass [kg]
Main Wing	66.1
Fuselage Pod	36.1
Horizontal Tail	6.5
Vertical Tail	12.7
Total Structural Mass	121.4

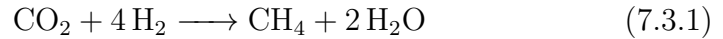
Table 7.3: Estimated Mass Breakdown

Equations were implemented in a MATLAB script and results are summarized in Table 7.3. The estimation is very satisfactory, since the total dry mass of the UAV is set at 180 kg; 120 kg of fuel bring the Maximum Takeoff Weight (MTOW) at 300 kg. This leaves plenty of margin for reinforcement and to implement the propulsion system, the fuel system, the avionics system and a basic landing system; all but the latter will be discussed in this chapter.

7.3 Propulsion System

The propulsion system of the UAV is built around a single chemical thruster. Thanks to the low aerodynamic drag, during the cruise phase the thrust requirement is fairly low, allowing the use of a small nozzle; in this case the *NASA/MSFC RCS Engine* was selected, a small attitude control thruster which uses liquid CH_4 as fuel and LOX as the oxidizer. The choice of this propulsion system gives several advantages [23]:

- **Constant Performance:** since both the fuel and the oxidizer are stored onboard, there is no effect of the altitude or external conditions on the thrust generated.
- **Compact System:** with this configuration, most of the propulsion system is stored inside the fuselage, protected from external conditions and minimizing drag.
- **Availability of Resources:** both the liquid CH_4 and the liquid oxygen could be produced in situ, thus allowing for an almost unlimited supply of fuel, to fully exploit the capability to reuse of the UAV [1]. The production cycle relies on the Sabatier Process, which has been already successfully demonstrated on the ISS [26, 27]:
 1. Carbon Dioxide Collection: Atmospheric CO_2 is collected and compressed.
 2. Sabatier Reaction: CO_2 reacts with hydrogen (extracted from ground ice) to produce methane and water:



3. Electrolysis: The water byproduct is split to generate oxygen (for the oxidizer or for life-support systems) and recycle the hydrogen [26, 27]:



Although this process is very energy-consuming, its closed-loop architecture allows the drone to refill its tanks at a ground station between long-range sorties.

The nozzle selected has a Specific Impulse $I_{sp} = 317s$, which translated in a Thrust Specific Fuel Consumption $TSFC = 3.217 \cdot 10^{-4} \text{Kg}/(\text{N} \cdot s)$ [23].

The 120 kg allocated for the fuel consists of both the fuel and the oxidizer.

7.4 Mission Payload

Since specific scientific objectives have not been defined for this conceptual study, the payload bay is designed with a modular architecture, allowing multiple instruments to be fitted in it, keeping in mind the 20 kg limit. For the baseline reconnaissance configuration, the following sensor package is proposed [1, 5]:

- **High-Resolution Optical Imager:** A visible-light camera equipped with telescopic optics. Its primary function is surface mapping to support ground activities.
- **LiDAR:** An active laser scanning system used to generate precise 3D Digital Terrain Models (DTM). Unlike passive cameras, LiDAR provides absolute distance measurements, essential for identifying safe landing zones or analyzing geological features.
- **Thermal Infrared Spectrometer:** A sensor capable of analyzing the thermal inertia of the ground. This is critical for detecting underground water-ice deposits, which retain heat differently than dry regolith.

The payload bay is located in the nose of the fuselage to maximize the field of view and balance the aircraft's center of gravity.

While propulsion is achieved chemically, all onboard systems, avionics, flight actuators, and the scientific payload rely on an electrical bus. Given the harsh thermal environment, standard commercial batteries performance would be insufficient; therefore, the design adopts Space-grade Lithium-Ion (Li-Ion) batteries and integrated heaters to maintain optimal operating temperatures. A conservative power budget was calculated considering the simultaneous operation of all on-board systems:

- **Estimated Average Load:** 150 W (continuous).
- **Allocated Battery Mass:** 5.0 kg .
- **Total Energy Capacity:** $\approx 1.25\text{ kWh}$ (assuming a pack specific energy of 250 Wh/kg).

This configuration provides an electrical endurance of approximately 8 hours; in comparison, the maximum endurance obtained during simulations was of under 5 hours, giving plenty of safety margin.

7.5 Guidance, Navigation, and Control

The primary challenge for navigation is the absence of a Global Navigation Satellite System (GNSS/GPS) on Mars. Relying solely on Inertial Measurement Units (IMUs) would lead to unacceptable position drift over the planned 1000 *km* flight. To correct this drift, the Flight Control Computer should utilize Terrain Relative Navigation (TRN) [31, 5]:

- **Visual Odometry:** Downward-facing cameras track high-contrast surface features (rocks, crater rims) to estimate velocity and position changes in real-time.
- **Map Matching:** The onboard computer compares live imagery with pre-loaded orbital maps (from MRO or Mars Express) to update its absolute position, ensuring the drone stays within the designated flight corridor.

7.6 Landing System

No definite landing system was developed for the UAV. To function properly, a landing gear would be too heavy to be worth it, and would require a lot of precision from the navigation computer to properly execute a landing procedure. Therefore, it is suggested to implement an inflatable bumper just under the fuselage, that would use a small amount of residual gas from the fuel tanks to inflate it and absorb the impact. It must be remembered, however, that thanks to several factors such as low gravity, lower mass at the end of the cruise and the ability to deploy flaps and reduce speed, landing forces are somewhat smaller.

More studies should be conducted to optimize this critical phase of the flight.

8 | Flight Performance Analysis

With the geometric configuration defined, the focus of the study shifts to the quantitative analysis of flight performance. To accurately predict the operational capabilities of the UAV, dedicated numerical simulations were developed in *MATLAB* [34].

8.1 XFLR5 Analysis Results

As a first step, the complete aircraft configuration, as seen in Figure 5.1, was analyzed in *XFLR5*; the results provide all the needed information about (C_L, C_D, C_m) that serve as the input for the performance calculations that will follow [35]. In this phase, it was studied a clean configuration of the UAV (no flaps deployed). Full results can be found in Appendix A.

8.1.1 Lift and Stall Characteristics

Figure 8.1 illustrates the lift curve ($C_L - \alpha$). The aircraft exhibits a linear and predictable behavior, until the maximum lift coefficient $C_{L_{max}} = 2.12$ at an angle of 16° . While *XFLR5* cannot accurately predict post-stall behavior, it is clear from the graph that the aircraft shows a sudden stall characteristic. In order to keep a safety margin, the aircraft should never be positioned at angles of attack close to 16° , to prevent a sudden gust from stalling the wing.

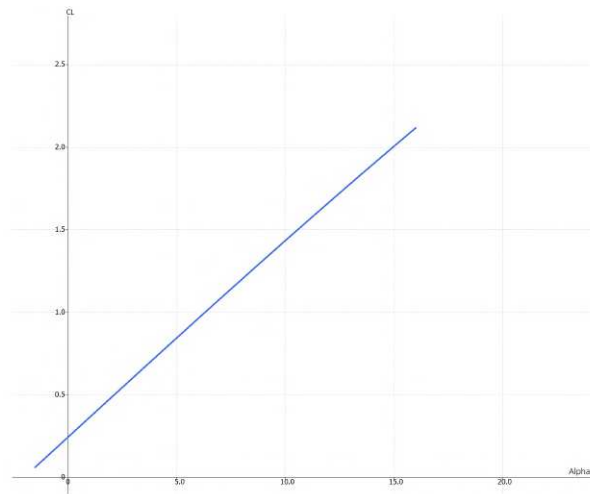


Figure 8.1: C_L vs α

8.1.2 Efficiency

The drag polar and the aerodynamic efficiency curve are critical metrics for this project, since they directly affect range and endurance capabilities.

It must be remembered, however, that the drag coefficient hereby presented relates only to flying surfaces, as neither the fuselage nor the twin booms are included in the XFLR5 geometry. Notably, the aerodynamic efficiency peaks at a specific angle of attack of 3° , with a value of 39.52, which is extremely high, even for an high AR aircraft. This operating point will be crucial during performance analysis, as it will be shown that it corresponds to the maximum endurance condition.

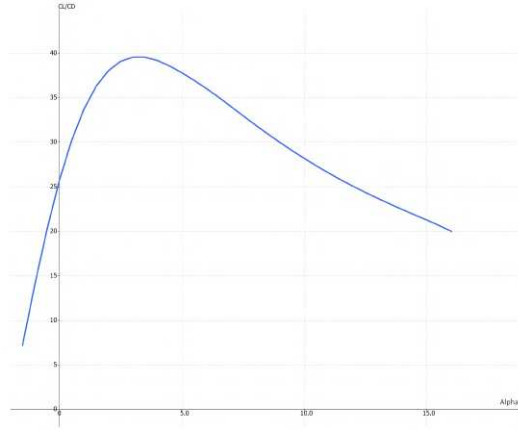


Figure 8.2: Aerodynamic Efficiency vs α

8.1.3 Aerodynamic Polar

Thanks to a MATLAB script provided with raw simulations data, the aerodynamic polar of the aircraft was computed. A generic polar $C_D = C_{D0} - K_1 C_L + K_2 C_L^m$ was calculated, obtaining the following coefficients: $C_{D0} = 0.0083$, $K_1 = 0.0006$, $K_2 = 0.0193$, $m = 2.13$. However, calculations proved that assuming a parabolic polar, thus neglecting K_1 and assuming $m = 2$, would result in an average error below 2%; for this reason, given the conceptual nature of this project, it was deemed a valid simplification.

The final aerodynamic polar is therefore: $C_D = C_{D0} + K C_L^m = 0.0083 + 0.01931 \cdot C_L^2$

8.1.4 High-Lift Devices Analysis

As indicated in Table 6.1, the main wing of the UAV is fitted with plain flaps on its trailing edge; these high-lift devices extend over the aft 20% of the chord and are located only on the rectangular portion of the wing. No leading-edge devices are installed.

Simulations were performed with flap deflections of $\delta_f = 5^\circ$ and $\delta_f = 10^\circ$. Full results are available in Appendix A. The results, presented in Figure 8.3, show the aerodynamic response of the aircraft under these configurations.

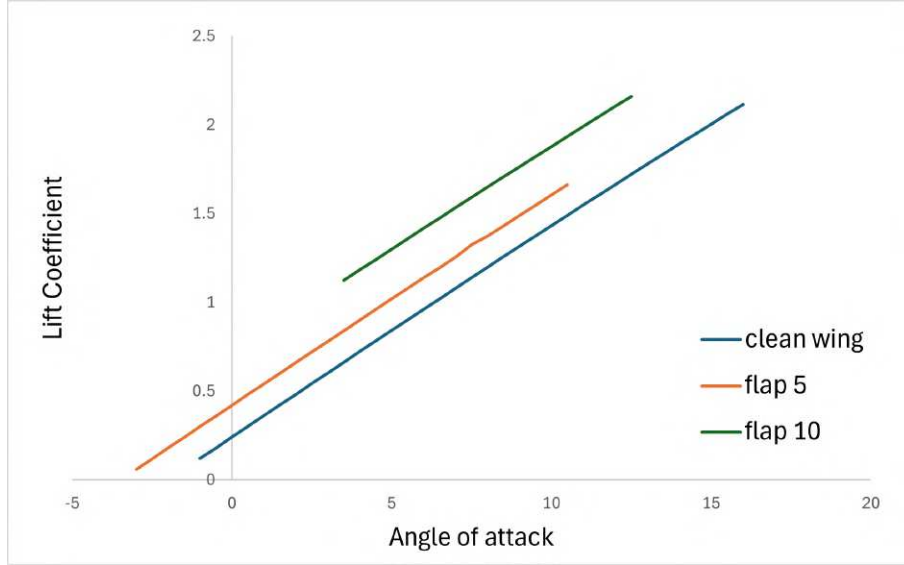


Figure 8.3: $C_L - \alpha$ curve at various flap configurations

As expected from theory, the lift curve slope ($C_{L\alpha}$) remains practically unchanged, while the zero-lift angle of attack ($\alpha_{L=0}$) shifts towards lower values [4]. It is important to observe the effect of flaps on the stall angle, as this represents a significant safety constraint for the mission. As shown in the graph, the lift curves for the deployed configurations terminate at lower angles of attack compared to the clean wing. Specifically, the 10° configuration exhibits a significantly reduced operational alpha range. This happens because the deflection of trailing-edge flaps increases the airfoil camber, which in turn intensifies the adverse pressure gradient on the suction side, which causes the boundary layer to separate earlier compared to the clean configuration [4]. Consequently, while the maximum lift coefficient ($C_{L_{max}}$) increases, the stall angle decreases, and aerodynamic drag rises significantly.

This reduction in the stall angle imposes a critical operational constraint: the flight control system must ensure that, when flaps are deployed, the pitch attitude is strictly maintained within this reduced alpha range to prevent premature stall. For this reason, the use of flaps will be restricted to the landing sequence (which is not covered in this document). For all other flight phases, a clean wing configuration will be adopted to maximize aerodynamic efficiency and ensure a wider margin of safety against stall.

8.2 Cruise Programs and Propulsive Model

Four different cruise programs were studied, in order to allow for maximum data gathering and flexibility of operation:

- Constant TAS program;
- Constant thrust (T) program;
- Constant C_L program;
- Constant TAS and constant C_L program (Breguet cruise program).

Moreover, optimization points were researched for each program.

The fundamental equations for the cruise conditions are:

$$\begin{cases} L = W = m(t) \cdot g_M \\ T = D \end{cases} \quad (8.2.1)$$

Where $m(t)$ is the instantaneous mass of the aircraft [kg], which decreases over time as fuel is consumed. Since the propulsion is provided by a rocket thruster, the fuel consumption is directly proportional to the thrust generated; the Specific Fuel Consumption is defined as $c = \frac{1}{I_{sp} g_0}$. Thus, the mass flow rate is $\dot{m} = -cT$ [23]. From this fundamental equation, the following program performances were derived.

8.2.1 Constant TAS Program

In this program, the True Airspeed (TAS) is held constant throughout the cruise phase. As fuel is consumed and mass decreases, the lift required to maintain level flight decreases as well. Consequently, the flight computer must progressively reduce the angle of attack, thereby reducing the drag coefficient over time.

Using the parabolic drag polar approximation, the drag force can be expressed as a function of instantaneous mass:

$$D(m) = A + Bm^2 \quad (8.2.2)$$

Where the constant coefficients are defined as:

$$A = \frac{1}{2} \rho V^2 S C_{D0} \quad \text{and} \quad B = \frac{2k g_M^2}{\rho T A S^2 S} \quad (8.2.3)$$

The differential equation for time derives from the mass flow relationship $\frac{dm}{dt} = -cD(m)$:

$$dt = -\frac{dm}{c(A + Bm^2)} \quad (8.2.4)$$

Integrating this expression from the initial mass m_i to the final mass m_f , and managing the negative sign by swapping the integration limits, yields an analytical solution based on the arctangent function:

$$\Delta t = \frac{1}{c\sqrt{AB}} \left[\arctan \left(\sqrt{\frac{B}{A}} m_i \right) - \arctan \left(\sqrt{\frac{B}{A}} m_f \right) \right] \quad (8.2.5)$$

The total range is simply the constant velocity multiplied by the endurance:

$$R = V \cdot \Delta t \quad (8.2.6)$$

To maximize performance in this mode, the cruise velocity must be carefully selected. Ideally, one should choose the velocity corresponding to the maximum aerodynamic efficiency (E_{max}). However, since the optimal velocity changes as weight drops, a fixed speed must be selected based on a reference weight, for example the initial weight or the mean weight:

$$TAS_{opt} = \sqrt{\frac{2 \cdot W_{ref}}{\rho S C_{L,Emax}}} \quad (8.2.7)$$

8.2.2 Constant Thrust Program

In this cruise program, the engine thrust is fixed at a specific setting ($T = \text{constant}$). Consequently, the mass flow rate is constant, and the mass decreases linearly with time: $m(t) = m_i - cTt$. Since $T = D$, as the weight decreases, the aircraft accelerates to find a new equilibrium. The flight velocity V is constrained by the equation:

$$T = \frac{1}{2} \rho T A S^2 S C_{D0} + \frac{2k(mg_M)^2}{\rho T A S^2 S} \quad (8.2.8)$$

This leads to an equation in TAS , which solved yields:

$$TAS(m) = \sqrt{\frac{T + \sqrt{T^2 - 4GHm^2}}{2G}} \quad (8.2.9)$$

Where the constant terms are $G = \frac{1}{2} \rho S C_{D0}$ and $H = \frac{2kg_M^2}{\rho S}$. The range is calculated by numerically integrating this velocity profile over the mass interval:

$$R = \int_{m_i}^{m_f} TAS(m) \frac{dt}{dm} dm = \frac{1}{cT} \int_{m_f}^{m_i} \sqrt{\frac{T + \sqrt{T^2 - 4GHm^2}}{2G}} dm \quad (8.2.10)$$

It is not possible to determine a maximum efficiency condition for this program, which, as will be shown in the results, is the most inefficient.

8.2.3 Constant C_L Program

In this program, the aircraft maintains a fixed attitude, implying constant C_L and constant aerodynamic efficiency $E = C_L/C_D$. Since $L = W$, as the mass $m(t)$ decreases, the velocity must decrease according to $V(t) \propto \sqrt{m(t)}$. The thrust required is thus $T = W/E$. Substituting into the mass flow equation and integrating, the total endurance is obtained:

$$\Delta t = \int_{m_i}^{m_f} dt = \frac{E}{cg_M} \ln \left(\frac{m_i}{m_f} \right) \quad (8.2.11)$$

Thus, it is quite clear to spot that it is possible to achieve maximum endurance by flying at the maximum efficiency lift coefficient (E_{max}).

The range is obtained by integrating $V(t)dt$, which yields:

$$R = \frac{2}{cg_M} \sqrt{\frac{2g_M}{\rho S C_L}} \frac{C_L}{C_D} (\sqrt{m_i} - \sqrt{m_f}) \quad (8.2.12)$$

Optimizing this equation reveals that maximizing range requires maximizing the term $\frac{\sqrt{C_L}}{C_D}$, which corresponds to a lift coefficient $C_{L,opt} = \sqrt{\frac{C_{D0}}{3k}}$.

8.2.4 Breguet Cruise Program

This strategy assumes that both the true airspeed and the lift coefficient are held constant throughout the flight. Since C_L is constant, the drag coefficient C_D (and thus the aerodynamic efficiency $E = C_L/C_D$) remains constant as well.

Starting from the drag equation:

$$D = \frac{1}{2} \rho T A S^2 S C_D \quad (8.2.13)$$

And substituting density from the lift equation ($L = mg_M = \frac{1}{2} \rho T A S^2 S C_L$), we can express drag as a linear function of mass:

$$D(m) = mg_M \frac{C_D}{C_L} = \lambda m \quad \text{where } \lambda = g_M \frac{C_D}{C_L} \quad (8.2.14)$$

Substituting this into the mass flow equation ($\dot{m} = -cD$):

$$\dot{m} = -c\lambda m = -\alpha m \quad \text{where } \alpha = c\lambda \quad (8.2.15)$$

This is a simple first-order differential equation. Integrating yields an exponential mass decay:

$$m(t) = m_i e^{-\alpha t} \quad (8.2.16)$$

Solving for the flight time by setting $m(t) = m_f$:

$$\Delta t = \frac{1}{\alpha} \ln \left(\frac{m_i}{m_f} \right) \quad (8.2.17)$$

The range is obtained by multiplying the constant velocity by the endurance:

$$\Delta x = V \cdot \Delta t = \frac{V}{\alpha} \ln \left(\frac{m_i}{m_f} \right) \quad (8.2.18)$$

Altitude Variation

A critical consequence of maintaining constant TAS and C_L while mass decreases is that the required atmospheric density must also decrease, which implies that the aircraft continuously climbs. Using the exponential atmospheric model $\rho(h) = \rho_0 e^{-h/H}$ (with scale height $H = 10700 \text{ m}$):

$$h(m) = -H \ln \left(\frac{2mg_M}{\rho_0 T A S^2 S C_L} \right) \quad (8.2.19)$$

Calculating the difference between the final and initial altitude, we obtain a relation independent of velocity and aerodynamics, solely dependent on the mass ratio:

$$\Delta h = H \ln \left(\frac{m_i}{m_f} \right) \quad (8.2.20)$$

This confirms that the optimal strategy for a fixed-geometry cruise naturally results in a "drift up" flight profile, thus violating mission requirements.

8.3 Optimization of Cruise Programs

Having defined the governing equations for range and endurance, optimization criteria were derived to identify the ideal operating conditions for each program, to maximize the UAV's performance.

8.3.1 Constant C_L Program Optimization

The optimization goal is to find the specific C_L (and thus α) that maximizes performance. For endurance optimization, recalling that $\Delta t \propto \frac{C_L}{C_D}$, the condition is simply:

$$\frac{d}{dC_L} \left(\frac{C_L}{C_{D0} + kC_L^2} \right) = 0 \implies C_{L,opt,end} = C_{L_{E_{max}}} \quad (8.3.1)$$

This corresponds exactly to the condition for maximum aerodynamic efficiency (E_{max}).

For range optimization, with a similar procedure it can be found that:

$$C_{L,opt,range} = \frac{1}{\sqrt{3}} \cdot C_{L_{E_{max}}} \quad (8.3.2)$$

This result is notable: it implies that to maximize the range, the aircraft must fly faster and at a lower angle of attack compared to the maximum efficiency condition.

8.3.2 Constant TAS Program Optimization

The optimization seeks the specific velocity V_{opt} that maximizes performance; maximizing the endurance function is equivalent to finding the velocity that minimizes the drag at a representative weight. Mathematically, this leads to the condition where the zero-lift drag equals the induced drag ($A = B$), which implies flying at the velocity for maximum efficiency. Consequently, $TAS_{E_{max}}$ yields the maximum endurance. The range is simply $TAS \cdot \Delta t$.

8.3.3 Breguet Program Optimization

The optimization follows the same logic as the constant C_L case, but with a different result due to the additional constraint regarding the speed. However, it can be computed that both maximum endurance and maximum range can be achieved by flying at the maximum efficiency condition.

8.3.4 Constant Thrust Program Optimization

The analytical optimization for the constant thrust strategy is algebraically complex; however, it can be found that the optimal thrust setting T^* is the one that allows the aircraft to fly as close as possible to the condition of maximum efficiency (E_{max}) throughout the cruise. Since the velocity changes as mass decreases, no single thrust setting can maintain the aircraft perfectly at E_{max} . The function $\Delta t(T)$ is monotonically decreasing with thrust (higher thrust burns fuel faster), while range presents a peak where the thrust balances drag at the optimal cruise speed. This is a highly inefficient cruise program.

8.4 Simulation Results and Discussion

The numerical simulation was performed for three distinct flight attitudes, defined by their initial angle of attack α . These configurations correspond to specific operating points identified during the aerodynamic polar analysis:

- **Configuration A** ($\alpha = 1^\circ$): Represents the free-controls trim position ($C_M = 0$). In this attitude, the aircraft theoretically maintains stable flight without requiring control surface deflection.
- **Configuration B** ($\alpha = 3^\circ$): Corresponds to the condition of maximum aerodynamic efficiency (E_{max}), theoretically ideal for maximizing flight time.
- **Configuration C** ($\alpha = 0.88^\circ$): Represents the mathematical optimum for range derived analytically ($\frac{1}{\sqrt{3}}C_{L,E_{max}}$).

All these results refer to a clean-wing configuration (flaps retracted). Alternative configurations deploying various flap settings were also simulated but consistently showed inferior performance. The aggregate results are presented in Figure 8.4 and Figure 8.5. The analysis of this data leads to several crucial conclusions regarding the ideal cruise strategy.

8.4.1 Comparison of Cruise Programs

Looking at both figures, the simulation unequivocally identifies the constant lift coefficient program as the superior one. Across all configurations, maintaining a constant C_L (and thus constant L/D) shows significantly better results compared to maintaining constant thrust or constant TAS. For example, in Configuration A, the Constant C_L strategy achieves a maximum range of 1594 km , whereas the constant thrust program achieves only 1097 km (-31%) and the constant TAS program reaches 1521 km .

This performance gap is physically justified: by flying at constant C_L , the aircraft continuously adjusts its speed as its weight decreases due to fuel burn ($V \propto \sqrt{W}$), thus ensuring that the wing always operates at its most efficient point for that specific weight. In contrast, the constant TAS program forces the aircraft to pitch down to lower angles of attack as it gets lighter, moving away from the optimal aerodynamic efficiency.

The Breguet cruise program showed the worst results across all configurations, with ranges roughly half of the optimal strategy ($\approx 690 - 740\text{ km}$), confirming its unsuitability in addition to the operational issue of altitude gain ($\Delta h \approx 5465\text{ m}$).

8.4.2 Range vs. Endurance Trade-off

Comparing the three configurations, a distinct trade-off emerges between flight duration and distance covered. Figure 8.5 shows that Configuration B delivers the longest flight time, reaching a peak of 4.67 hours with the Constant C_L cruise program. This confirms the analytical prediction that flying at E_{max} minimizes the

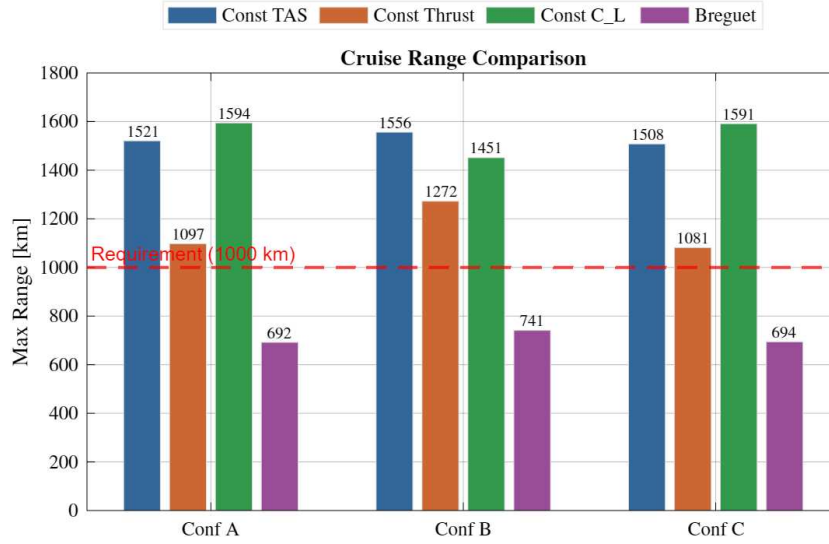


Figure 8.4: Comparison of maximum range

thrust required to stay airborne, thus minimizing fuel consumption. However, Figure 8.4 and Figure 8.5 highlight that endurance does not equal range. Due to the lower flight speed associated with the high angle of attack of Configuration B, the total distance covered is limited to 1451 *km*, which is still far more than the required range.

Configuration A proves that flying faster and at a lower angle of attack is beneficial for distance. Despite having a shorter endurance (3.98 hours), the higher average velocity allows it to cover 1594 *km*, outperforming Configuration B by over 140 *km*. It must be pointed out, however, that despite Configuration C theoretically being the one that would allow for maximum range, it is actually slightly outperformed by Configuration A by a mere 3 *km*. This is most probably due to the fact that both configurations fly at a very similar angle of attack, so it is hard to notice a significant difference.

8.4.3 Mission Applications

The primary mission requirement, set in Chapter 5, is long-range reconnaissance with a target of at least 1000 *km*. Based on the results shown in Figure 8.4, Configuration A using the Constant C_L cruise program is selected as the baseline mission profile.

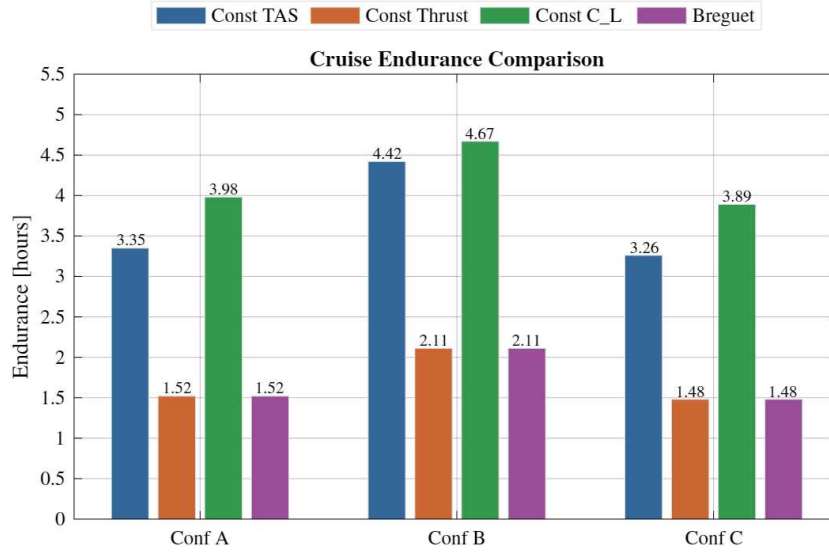


Figure 8.5: Comparison of maximum endurance

8.5 Climb Performance Analysis

As per the constructor's data sheets, the rocket engine described in Section 7.3 can deliver a maximum thrust of 66.7 N ; this, combined with the correct pitch, must give enough Rate of Climb (ROC) to safely compensate for any emergency situations. It must be noted, however, that the cruise altitude of 1000 m gives enough ground clearance to not make an excessive maneuver necessary.

The climb performance is governed by the excess power, defined as the difference between the power available from the engine and the power required to overcome drag.

A numerical simulation was performed to map the ROC across the entire lift coefficient envelope; the weight was set at 250 kg , with 50 kg of fuel already burned, to simulate a sudden in-flight emergency.

We define the Rate of Climb as:

$$ROC = \frac{TAS \cdot (T - D)}{W} \quad (8.5.1)$$

Results plotted in Figure 8.6 reveal a critical trade-off:

- $C_L < 0.35$ region: In this regime, corresponding to a high-speed cruise due to the low C_L , the ROC achievable is severely limited. Despite the low induced drag, the high TAS generates significant parasitic drag on all the surfaces. At these cruise speeds ($V > 110\text{ m/s}$), the total drag nearly equals the maximum thrust, leaving minimal excess power for climbing.
- $C_L \approx 0.725$ region: As the aircraft pitch is increased allowing it to slow down,

the parasitic drag drops. This creates a "sweet spot" where the sum of parasitic and induced drag is minimized. The simulation identifies an optimal lift coefficient of $C_L = 0.725$, allowing for a peak ROC of 1.963 m/s , which is the highest achievable value with this weight.

- $C_L > 1.4$ region: If the angle of attack is increased further, the induced drag rises exponentially, eventually overcoming the engine thrust.

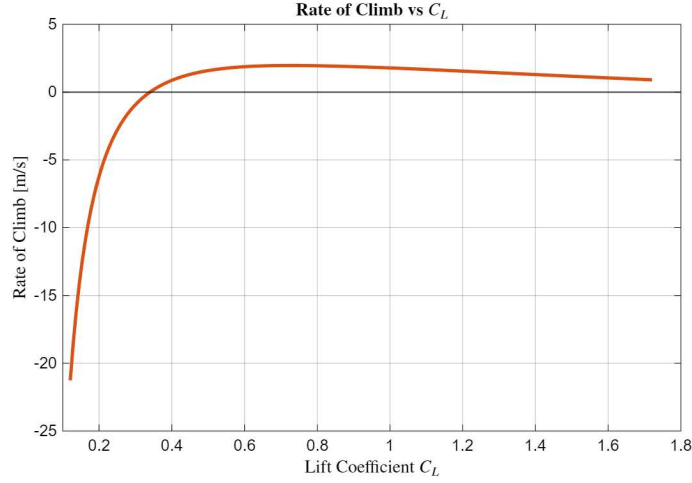


Figure 8.6: ROC vs C_L at maximum thrust

Thus, in order to initiate an efficient climb starting from cruise conditions, the flight control system must not only apply full throttle but also pitch up to decelerate the aircraft, settling at an angle of attack $\alpha \approx 4^\circ$

8.6 Gliding Performance Analysis

In the event of an engine failure or during the descent phase when all the available fuel has already been burned up, the UAV operates as a glider with a mass of 180 kg . The gliding performance is dictated solely by the lift-to-drag ratio (E), which must be maximized in order to extend the range to its maximum.

The key metrics are the glide angle (γ) and the sink rate (V_z):

$$\tan(\gamma) = \frac{1}{E} \quad \text{and} \quad V_z = V \sin(\gamma) \approx \frac{V}{E} \quad (8.6.1)$$

Thanks to the high aspect ratio design and the clean aerodynamic configuration, the UAV achieves a maximum glide ratio of approximately 39.5:1, meaning that it will travel for 39.5 m in the horizontal direction for every meter lost in altitude. This has profound operational implications: from a cruise altitude of 1000 m , the aircraft can glide for approximately 39 km before touching the ground.

This exceptional gliding capability provides a massive safety buffer, and allows the flight computer ample time to identify a safe landing spot or reach a recovery zone even in case of total propulsion loss. Moreover, gliding endurance can be maximized

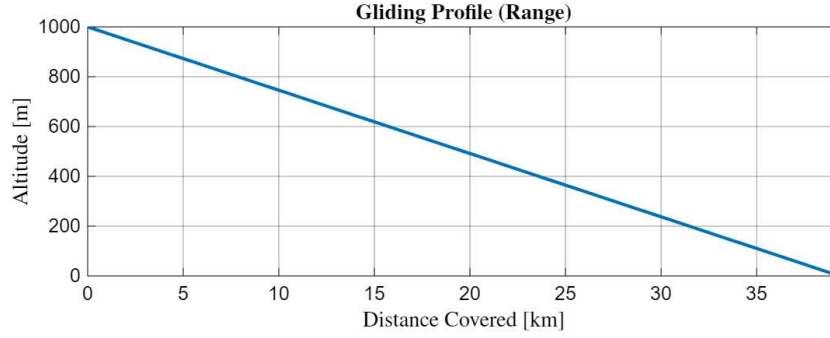


Figure 8.7: Gliding range from cruise altitude

by flying at $\sqrt{3} \cdot C_{L_{Emax}}$, resulting in a slightly longer air time, at the expenses of reduced range; results are shown in Figure 8.7 and Figure 8.8. Furthermore, it is pointed out that, while atmospheric density does increase while losing altitude, this factor plays a marginal role in the overall performance, but it was still accounted for in the simulation.

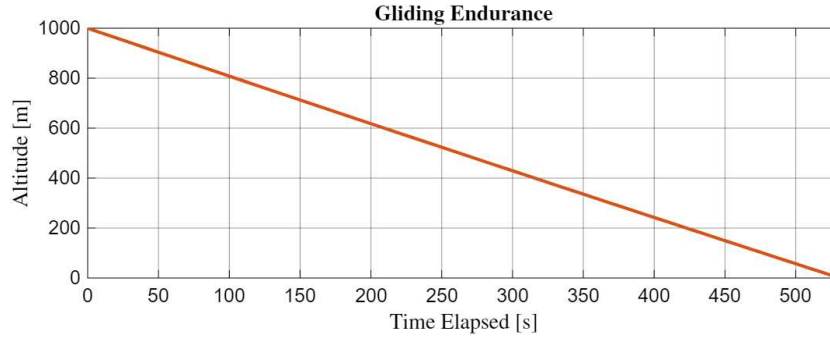


Figure 8.8: Gliding endurance from cruise altitude

8.7 Turning Performance Analysis

Maneuverability on Mars presents unique challenges compared to Earth. The combination of high airspeed and low gravity results in significantly larger turn radii for the same bank angle. The turn radius is defined as:

$$R = \frac{TAS^2}{g_M \cdot \tan(\mu)} \quad (8.7.1)$$

Where μ is the bank angle. The only turn strategy that was studied was the coordinated turn, during which the aircraft keeps constant altitude, airspeed and turning radius, without any sideslip β . The coordinated turn was deemed the only useful turn strategy, while also being easier to manage for the flight control system. In this case, the simulation analyzed the UAV having a mass of 240 kg , thus having burned half of its fuel, simulating a 180° turn to head back to base, while having some spare fuel on board for the turn itself and for safety. Simulations results show that for a bank angle of $\mu = 30^\circ$, the turn radius is approximately 3500 m . Even banking aggressively to $\mu = 45^\circ$ only reduces the radius to $\approx 2000\text{ m}$. A graphic comparison is shown in Figure 8.9

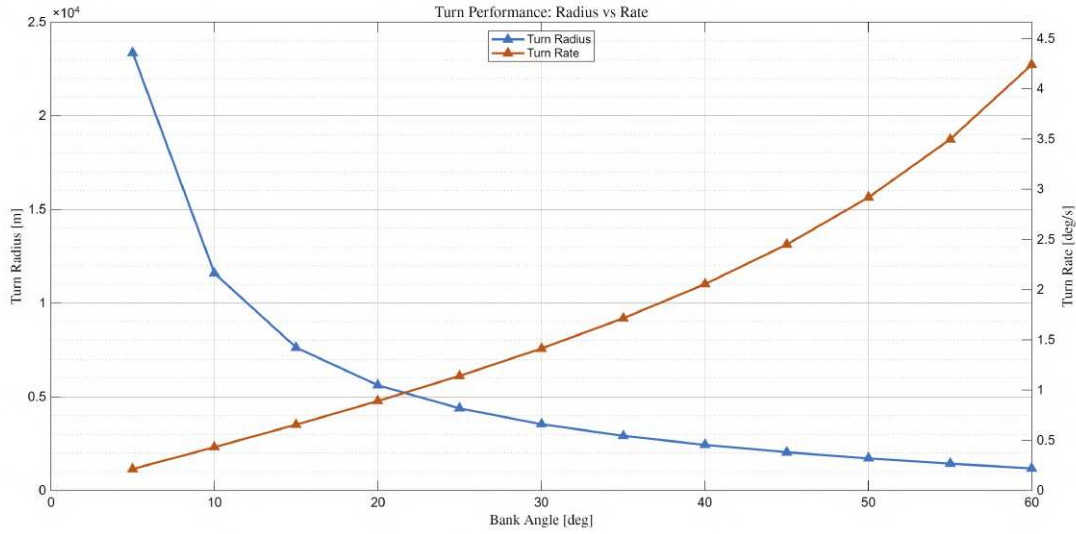


Figure 8.9: Turn performance

Bank Angle	Radius	Turn Rate	Time	Load Factor	Req. C_L
μ [°]	R [m]	Ω [°/s]	t_{turn} [s]	n [-]	$C_{L,req}$ [-]
5	23358	0.21	840.5	1.00	0.605
10	11589	0.43	417.0	1.02	0.612
15	7627	0.66	274.4	1.04	0.624
20	5615	0.89	202.0	1.06	0.641
25	4382	1.14	157.7	1.10	0.665
30	3539	1.41	127.4	1.16	0.696
35	2918	1.71	105.0	1.22	0.736
40	2435	2.05	87.6	1.31	0.787
45	2044	2.45	73.5	1.41	0.852
50	1715	2.92	61.7	1.56	0.938
55	1431	3.50	51.5	1.74	1.051
60	1180	4.24	42.5	2.00	1.206

Table 8.1: Turn performance as a function of bank angle (μ) for a 180° coordinated turn

This calculation highlights a critical operational constraint: the aircraft requires several kilometers of lateral space to complete a standard turn. Tighter turns are technically possible by increasing the bank angle, but some limitations must be considered:

- **Load Factor:** $n = 1/\cos \mu$ increases stress on the long wings. The load factor is therefore limited by the flight computer at $n = 2$, resulting in a bank angle of $\mu = 60^\circ$.
- **Stall Speed:** It increases as $V_{stall,turn} = V_{stall}\sqrt{n}$.
- **Induced Drag:** A steep turn bleeds energy rapidly, which the small rocket engine may not be able to compensate for, possibly forcing a spiraling descent.

Therefore, mission planning must account for these wide turning circles, avoiding too tight maneuvers. For a more clear overview, Table 8.1 details the exact numerical values for the turn performance across the bank angle envelope.

9 | The Takeoff Phase

As was already mentioned, one of the requirements for this project is for the UAV to be reusable, and not limited to a single flight when dropped from altitude. For this reason, a small landing strip must be constructed in order for the aircraft to take off and land safely.

In order not to compromise the UAV's performance once at cruising altitude due to the high amount of fuel burned to lift itself from the ground, the takeoff phase will be propelled by an External Takeoff Rig (ETR), itself being a reusable system composed of several key components:

- **Structural Subsystem:** very simple and lightweight, its main purpose is to hold together all the other components of the ETR and to connect to the UAV's fuselage. This subsystem must allow for a quick separation from the UAV once cruising altitude is reached.
- **Propulsion and Fuel Subsystems:** since the aircraft at takeoff will have a mass of 300 kg when fully fueled, a more powerful nozzle will be required to accelerate the UAV+ETR system up to a sufficient speed at which enough lift is generated. The engine deemed suitable for this task is the *VR900* by *Intuitive Machines*: it is a flight-proven design, used on the *Nova-C* lunar lander in 2024, capable of a maximum thrust of 3100 N while having a mass of under 10 kg , thanks to its construction by additive manufacturing. Moreover, the nozzle is pressure-fed, so it will not need turbopumps [37]. For compatibility reason, this propulsive system will be also fueled by liquid CH_4 and LOX , as the primary nozzle on the drone.
- **Skid:** a conventional landing gear could easily account for up to 10% of an aircraft's dry mass, while being a dead weight during the cruise phase [2]. Since it is needed for the UAV to stay aloft as long as possible, a landing gear is simply not worth its weight. In order to allow for a smooth takeoff run, however, the UAV+ETR system will be mounted at an angle on a detachable skid, which will allow for the system to accelerate while sliding on the ground. The skid cannot be made of the same composite materials as the UAV, since the Martian regolith will quickly destroy it, but instead of a metallic but lightweight material, such as aluminum. For the landing sequence, the UAV will simply deploy maximum flaps and stall itself a few meters above the ground; thanks to Mars' low gravity, impact forces are much lower compared to what we experience on Earth.

- **Recovery Subsystem:** as mentioned, the External Takeoff Rig is meant to be reusable itself with minimum maintenance between takeoffs. For this reason, the rig is fitted with a small parachute which will stabilize and slow down its descent until a final landing burn can be performed by the engine [37].
- **Guidance, Navigation and Control Subsystem:** the takeoff maneuver in itself is fairly easy, with all the stability corrections and lift generation taken care of by the UAV, leaving just a few basic operations to be performed by the ETR alone, mainly during its landing sequence.

9.1 Takeoff Sequence

In order to determine the appropriate allocation of mass for the dry weight of the ETR and the amount of fuel needed to bring the UAV up to cruising altitude and speed, an iterative calculation was performed until the hereby presented results were obtained.

Masses where determined as follows:

Component	Mass
UAV MTOW	300.0 kg
ETR Dry Weight	26.5 kg
$m_{fuel,takeoff}$	22.7 kg
Total MTOW	349.2 kg
Skid Mass	15.0 kg
Phase 1 Total Mass	364.2 kg

Table 9.1: Mass Breakdown

Three primary constraints were imposed for this maneuver:

- Maximum acceleration $a_{max} = 1.5 \cdot g_M = 5.58 \frac{m}{s^2}$ to avoid excessive structural stresses;
- Maximum velocity $V_{max} = 140 m/s$ to keep an adequate margin against compressibility effects;
- Constant angle of attack $\alpha = 13^\circ$.

The takeoff sequence is divided in four phases, characterized as follows:

- **Phase 1:** the ETR nozzle is ignited and the UAV+ETR+Skid system begins to accelerate, sliding on the landing strip. Since the UAV is already mounted

on the skid at a $\alpha = 13^\circ$ angle of attack, it will not deploy its flaps, but it will still generate a coefficient of lift $C_L = 1.778$. The necessary lift to make the total MTOW takeoff is obtained once the velocity $TAS = 58.59 m/s$ is reached which, given the constant acceleration, takes less than 11 seconds; at this point the skid detaches and the system can begin to climb, keeping the angle of attack constant. Note how friction with the ground and other dissipative effects have been ignored, since this is just a preliminary iteration of the concept.

Moreover, it is possible to estimate the length of the required runway to be around 310 m.

- **Phase 2:** the ETR+UAV system continues to accelerate at a_{max} until V_{max} is reached. This is predicted to happen at $\approx 330 m$ of altitude.
- **Phase 3:** the climb is now performed at constant V_{max} speed until the desired cruising altitude is reached. At this point, the ETR detaches from the UAV, which is free to begin its mission, and begins Phase 4 as a stand-alone system.

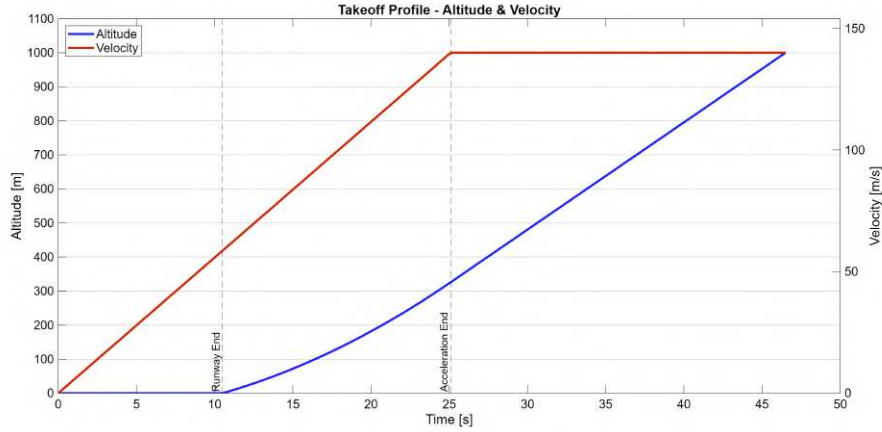


Figure 9.1: Takeoff Profile

- **Phase 4:** thanks to some residual speed and the low Martian gravity, the ETR continues to climb until it reaches 1100 m of altitude. It then opens its front-mounted parachute, which will help to stabilize and slow down its descent to around 30 m/s, and correctly orienting the ETR with the nozzle pointing downwards. Once 30 m of altitude have been reached, the nozzle fires up once again for less than 2 seconds, allowing an almost zero vertical velocity touchdown for the External Takeoff Rig.

A brief summary of these phases is presented in Table 9.2.

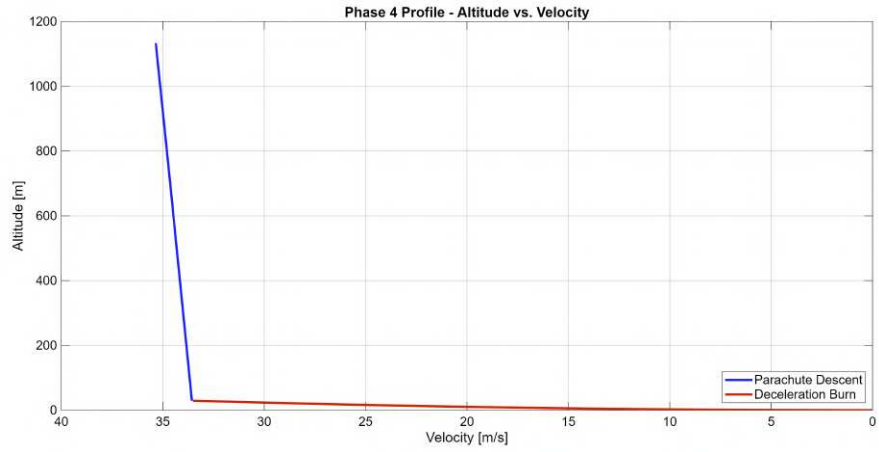


Figure 9.2: Phase 4 Breakdown

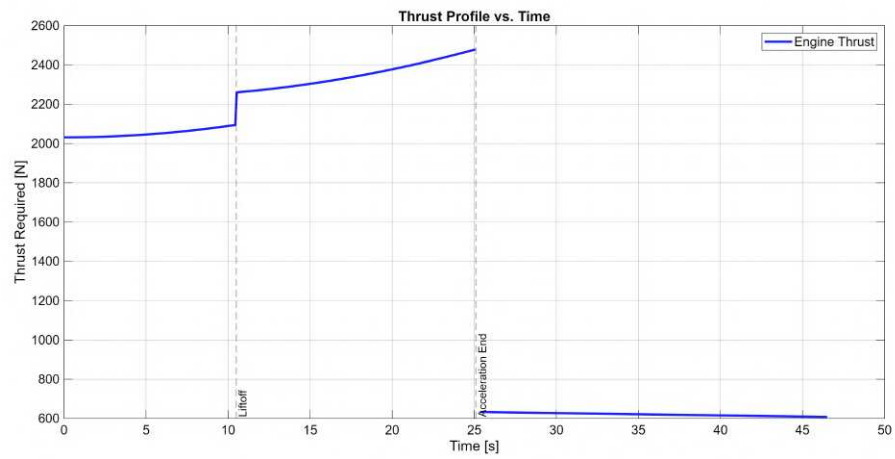


Figure 9.3: Takeoff Phase Thrust Profile

Phase	Initial Mass	Final Mass	Fuel Burned	Phase Duration
Units	$[kg]$	$[kg]$	$[kg]$	$[s]$
Phase 1:	364.2	342.2	7	10.5
Phase 2:	342.2	330.8	11.4	14.6
Phase 3:	330.8	326.5	4.3	21.4
Phase 4:	326.5	26.16	0.34	43

Table 9.2: Fuel Burn Breakdown and Time Estimations for Takeoff Phases

10 | Gust Sensitivity

Before discussing in more detail the stability analysis, it is crucial to address the expected behavior of the aircraft in the Martian atmosphere from a qualitative point of view. While the static atmospheric model used for performance calculations does not account for turbulence, the operational reality is far more dynamic and, potentially, hazardous.

10.1 Atmospheric Disturbances on Mars

The flight profile places the UAV within some sort of planetary boundary layer, where the interaction between the thin, cold atmosphere and the solar-heated surface could generate significant turbulence [1]. Two specific phenomena that may pose a threat to the airframe can be:

- **Thermal Vertical Gusts:** During the daytime, the heating of the surface gives birth to vertical currents similar to terrestrial thermals but often narrower and faster-rising. Large-Eddy Simulations (LES) indicate that these convective updrafts routinely reach vertical velocities of 10 m/s , with localized peaks exceeding $15 - 20\text{ m/s}$.
- **Turbulence due to Topography:** Since the mission involves reconnaissance also over craters and canyons, the wind interacting with these geological features could generate sharp and unpredictable wind currents.

Flight data reconstructed from the ExoMars 2016 Schiaparelli descent revealed wind speeds reaching peaks of 20 m/s . However, these maxima were identified in specific high-altitude shear layers (between 3.2 and 6 km) [12]. Standard Martian Climate Database (MCD) models predict lower values around 10 m/s near the surface. Consequently, this stability analysis adopts a disturbance magnitude equivalent to a $\approx 12\text{ m/s}$ gust. This value provides a safety margin over mean theoretical models, without over-dimensioning the aircraft for high-altitude shear phenomena that are outside the intended mission profile.

10.2 The Contribution of the Aspect Ratio

The peculiar geometry of the UAV, a high-aspect-ratio ($AR \approx 18.75$) wing with low wing loading ($W/S \approx 14\text{ kg/m}^2$), makes it particularly vulnerable to these disturbances [2]. From a flight mechanics perspective, the sensitivity to a vertical

gust w_g is driven by the change in the angle of attack [8]:

$$\Delta\alpha \approx \frac{w_g}{TAS} \quad (10.2.1)$$

Although the high airspeed will help to reduce $\Delta\alpha$ for a given gust intensity, the aerodynamic response of the airframe amplifies the effect:

- **Gust Load Factor:** The lift curve slope of a high-AR wing is steeper than that of a low-AR one. This means that even a small $\Delta\alpha$ induced by a gust translates into a large instantaneous angle of attack increment, resulting in a sudden vertical movement. Given the relatively low mass of the airframe, this could result in high vertical accelerations that stress the root structure.
- **Asymmetric Gust:** Having a wingspan of 20 m , it is probable that the aircraft will encounter a gust with some sort of span-wise gradient in its velocity, where one wingtip is inside a thermal updraft while the other is in still air. This could generate a massive rolling moment that the flight control system must be able to correct.

The stability analysis presented in the following section is a crucial aspect in determining the feasibility of this project, as it is a verification of the aircraft's ability to passively dampen these inevitable perturbations and survive the Martian weather. A more comprehensive aeroelastic analysis should be performed in the following stages of the design, in order to avoid the onset of other potentially dangerous aeroelastic issues, from an excessive twisting of the wing to the flutter phenomenon.

11 | Stability Analysis

In this section, the results of a more detailed stability analysis are reported and discussed. This analysis was performed once again by using the dedicated tool in XFLR5, in which the mass breakdown presented in Table 7.3 was used. Moreover, the payload's center of mass is located 0.5 m behind the nose of the fuselage, while the overall center of mass of the fuel system and of the fuel, is placed directly under the main wing, in the geometrical center of the fuselage. The longitudinal position of the center of gravity for the whole UAV was determined to be 0.8 m ahead of the leading edge of the main wing.

While a more dedicated software should be used in a following analysis to validate these results, they serve as a primary, fundamental tool to validate this conceptual design; the importance of stability analysis cannot be overstated, since no controlled flight could be possible without it, let alone in a peculiar atmospheric environment like the Martian one.

11.1 Introduction to Stability

When analyzing aircraft dynamics, stability is generally categorized into longitudinal and lateral-directional modes.

- **Longitudinal Stability:** it allows the aircraft to maintain the desired pitch attitude when subjected to a disturbance. To ensure static longitudinal stability, the slope of the pitching moment coefficient curve must be negative ($C_{m\alpha} < 0$), and the pitching moment coefficient at zero lift must be positive ($C_{m0} > 0$). Practically, this is achieved by making sure that the center of gravity (CoG) of the UAV is ahead of its neutral point (NP).

Graphically, this stability can be verified on the $C_m - \alpha$ plot by checking for a negative slope.

- **Lateral-Directional Stability:** Unlike longitudinal stability, which can often be analyzed in isolation, lateral and directional motions are coupled in flight dynamics. Therefore, stability requires the satisfaction of two simultaneous conditions [2]:

- **Directional Stability** ($C_{n\beta} > 0$): The aircraft must possess a positive yawing moment derivative. This can be interpreted as a stiffness, meaning that when the aircraft is subjected to a sideslip angle β , the vertical tail must generate a restoring moment that realigns the nose with the relative

wind. This explains the large vertical tail area in the proposed design showed in Chapter 6.

- **Lateral Stability ($C_{l\beta} < 0$):** The aircraft must possess a negative rolling moment coefficient. If the aircraft slips sideways, the differential lift on the wings that is created, must be able to generate a rolling moment that tends to level back the wings. This is usually achieved through the geometric dihedral angle (Γ).

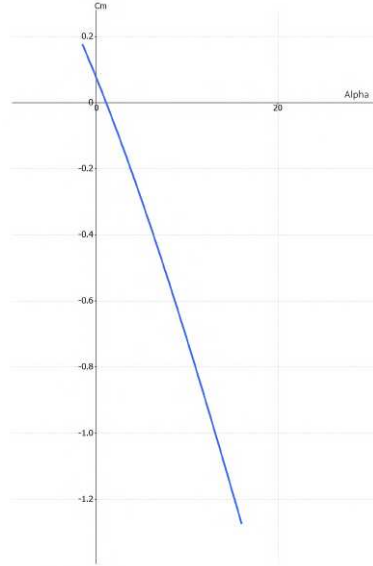


Figure 11.1: $C_m - \alpha$ plot for the clean wing configuration

11.2 Stability Analysis Results

The simulation was performed as described in the introduction of this Chapter, and gave satisfactory results, proving that the stability issues that affected initial designs for this UAV are now solved, and the proposed final design with the twin-boom configuration is indeed capable of a stable flight.

11.2.1 Longitudinal Stability Results

The simulation gives a pitching moment derivative of:

$$C_{m\alpha} = -4.037 \text{ rad}^{-1} \quad (11.2.1)$$

The negative sign confirms stability, as explained in the previous paragraph. Furthermore, the neutral point is found to be located at 1.523 m behind the aircraft nose tip; reminding that the UAV's center of gravity is located at 0.8 m from the

same reference point, the necessary condition for stability is satisfied. Moreover, the static margin is determined to be:

$$SM = \frac{X_{NP} - X_{CoG}}{\bar{c}} \approx 58.4\% \quad (11.2.2)$$

Which is an exceptionally large value.

11.2.2 Lateral-Directional Stability Results

Results confirm the effectiveness of the H-tail configuration as opposed to the single vertical fin of previous design iterations:

- **Directional:** $C_{n\beta} = +0.219$. The positive value indicates stability, crucial for keeping the UAV aligned with the flow.
- **Lateral:** $C_{l\beta} = -0.151$. The negative value confirms a stable dihedral effect.

11.3 Dynamic Stability Analysis

In addition to static equilibrium analysis, the dynamic response was also analyzed by solving the eigenvalues of the linearized equations of motion provided by the XFLR5 output files. It is reminded that a mode is considered stable if the real part of its eigenvalue is negative. Figure 11.2 illustrates the location of the aircraft's eigenvalues on the complex plane [2, 8]. The vertical axis shows the frequency of oscillation, while the horizontal axis represents the damping factor. The stability criterion requires that poles lie in the negative real part: it easy to spot that the poles for the critical oscillatory modes are all located to the left of the imaginary axis. Therefore, this confirms that the aircraft is dynamically stable and any oscillation will naturally decay over time.

11.3.1 Lateral-Directional Modes

The most critical aspect of the design was ensuring the stability of the coupled lateral modes, since early iterations of the design suffered from this issue. Analysis results showed that:

- **Dutch Roll:** The eigenvalues are $\lambda_{DR} = -0.1216 \pm 2.795i$; note that the real part is negative, as needed.
- **Spiral Mode:** The eigenvalue is $\lambda_{spiral} = 0.0024 + 0i$, in which the zero imaginary part confirms that this is a non-oscillatory motion, while the slightly

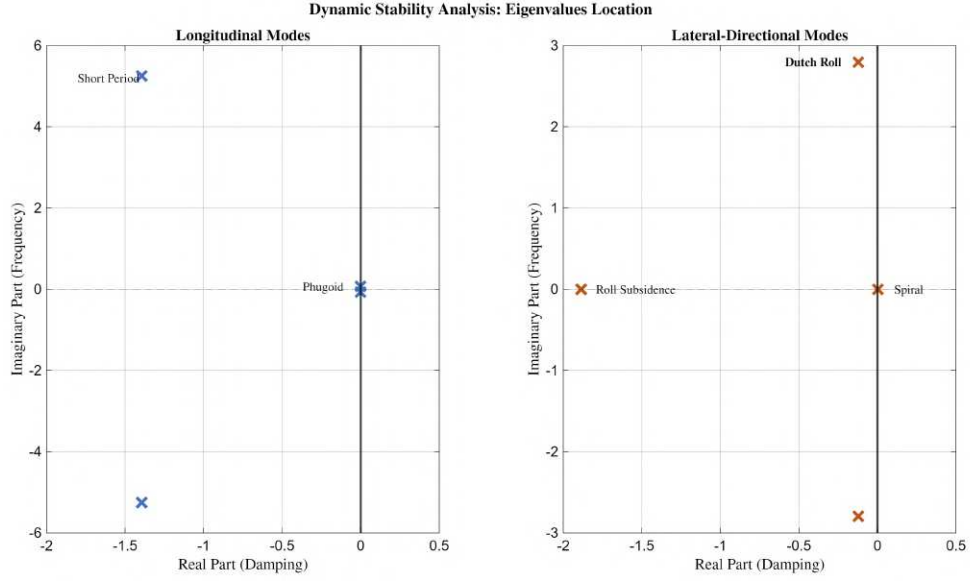


Figure 11.2: Root Locus of the aircraft's dynamic modes

positive real part indicates a slowly divergent trend. However, the time required to double the amplitude is significant (≈ 285 s). Such mild instability can still be considered acceptable, as the divergence is too slow to significantly affect handling qualities and it can be easily stabilized by the flight control system [6].

11.3.2 Dynamic Response Simulation

To further visualize the aircraft's behavior, a stability simulation was performed. The system was subjected to pulse disturbances in pitch and yaw, and the response shows that the oscillations are effectively damped, returning to equilibrium without diverging, confirming the airworthiness of the design [8].

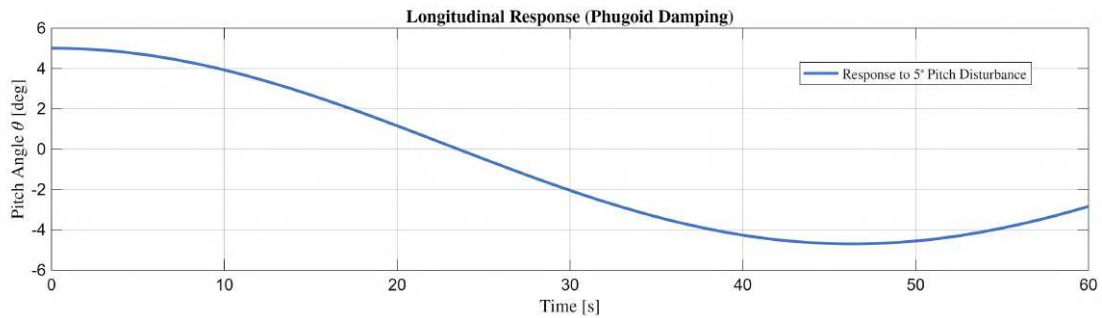


Figure 11.3: Phugoid mode

First, the system was subjected to a nose-up pitch attitude disturbance of $\theta_0 = 5^\circ$, simulating a sudden upwards gust of wind. This condition represents a post-disturbance state, equivalent to the attitude excursion induced by a vertical gust of

12 m/s while at nominal cruise speed, or an abrupt control input. The simulation verifies the aircraft's ability to passively return to equilibrium without active flight computer intervention.

A second analysis focused on the lateral-directional dynamics, subjecting the system to an initial yaw rate disturbance of 0.1 rad/s ($5.7^\circ/s$). This condition was selected to simulate the transient dynamic response triggered by a lateral gust of approximately 12 m/s hitting the aircraft during the cruise phase. This magnitude aligns with the environmental conditions assumed for the longitudinal analysis, representing a significant asymmetrical aerodynamic load acting on the vertical stabilizer. The orange curve tracks the yaw rate evolution over time. The simulation yields positive results: the coupled yaw-roll oscillations (Dutch Roll) are effectively damped within a few cycles. The amplitude drops significantly within the first 15 seconds, causing the system to return to equilibrium rapidly. This behavior confirms that the designed directional stability ($C_{n\beta}$) is sufficient to prevent dangerous divergent oscillations [2, 8].

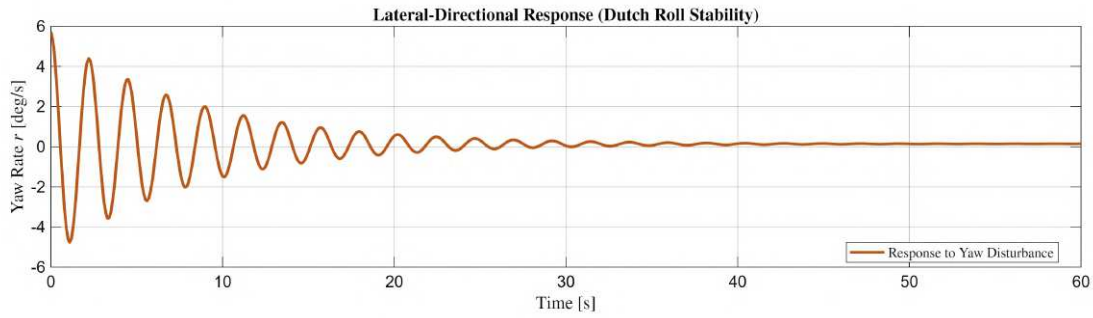


Figure 11.4: Dutch Roll stability

12 | The Flight Envelope

The definition of a flight envelope allows for a critical analysis and evaluation of the aircraft's capabilities and constraints, and, for the first time, they are all put together in a single, graphical presentation for a quick overlook. The flight envelope shows the limits, mainly of aerodynamic and structural nature of the UAV, therefore highlighting a safe area of operations, which can be crucial during the mission definition phase to design a safe flight [2].

For a more comprehensive analysis, two different flight envelopes were produced, with each one giving a unique point of view on the UAV's behavior:

- Altitude-Speed Flight Envelope;
- V-n Flight Envelope.

12.1 Altitude-Speed Flight Envelope

The $h-V$ diagram shows the allowed airspeed range across the operational altitudes up to the imposed ceiling of 2000 m . The "Safe Flight Zone", highlighted in a light blue color, is delimited by four constraints:

- **Stall Limit:** The left boundary, marked in a red line, represents the minimum speed required to sustain level flight, thus satisfying the fundamental relation $L = W$, with the curve also accounting for the variation of atmospheric density with altitude, according to the atmospheric model (Equation 3.3.2).
- **Thrust Limit:** The right boundary, marked in a blue line, represents the maximum speed achievable with the maximum available engine thrust of 65 N . It is the equilibrium point where total drag equals the maximum thrust.
- **Overspeed:** A hard constraint is set at $V_{NE} = 140\text{ m/s}$ to prevent compressibility effects and aeroelastic instability, with some margin.

The result is plotted in Figure 12.1.

One important consideration that must be made, is that at the nominal cruising altitude of 1000 m , the UAV shows a notable safe area of operation, giving plenty of margin against issues that could arise.

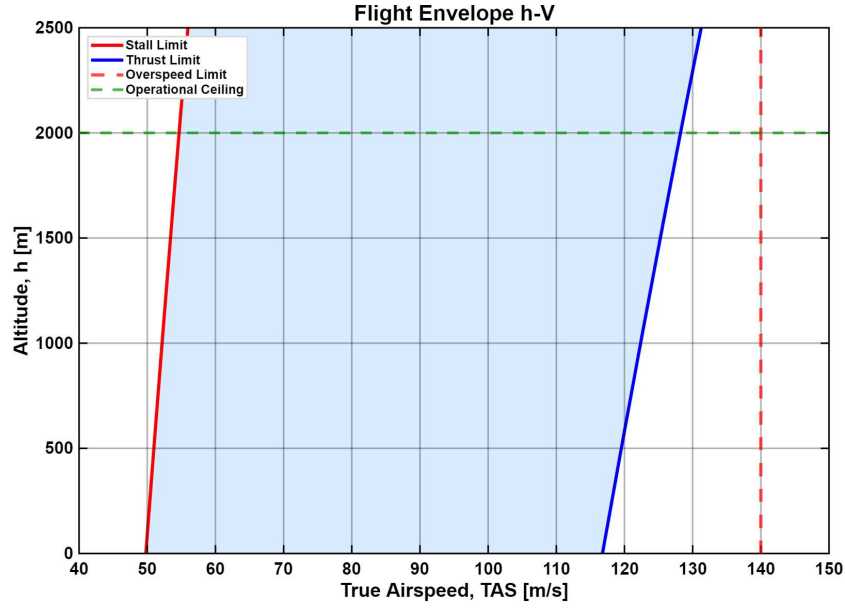


Figure 12.1: Altitude-Speed Flight Envelope

12.2 V-n Flight Envelope

The V-n flight envelope is more common and it also gives more important information; in particular, it can be useful for defining the limitations of the flight control system.

- **Structural Limits:** In order to minimize structural mass for the UAV, the limit load factors (indicated with black lines) are set to:

- Positive Limit: $n_{lim}^+ = +3.0 g$
- Negative Limit: $n_{lim}^- = -1.75 g$

- **Gust Envelope:** The pink lines represent the additional load factor induced by a vertical gust of wind of $15 m/s$; this value was chosen considering an average gust speed on Mars of $12 m/s$, so it also accounts for some safety margin. Given its high lift slope, the aircraft is pretty sensitive to turbulence. Despite this, the diagram highlights only one danger zone (in shaded red) at high speeds, where a $15 m/s$ gust could momentarily push the load factor beyond the $3g$ structural limit.

- **Autopilot Limit:** To ensure additional safety, an operational constraint is set at $n = 2.0 g$. This is the maximum load factor the autopilot is allowed to command during standard maneuvers. This limit is deemed reasonable, given the reconnaissance purpose of the aircraft.

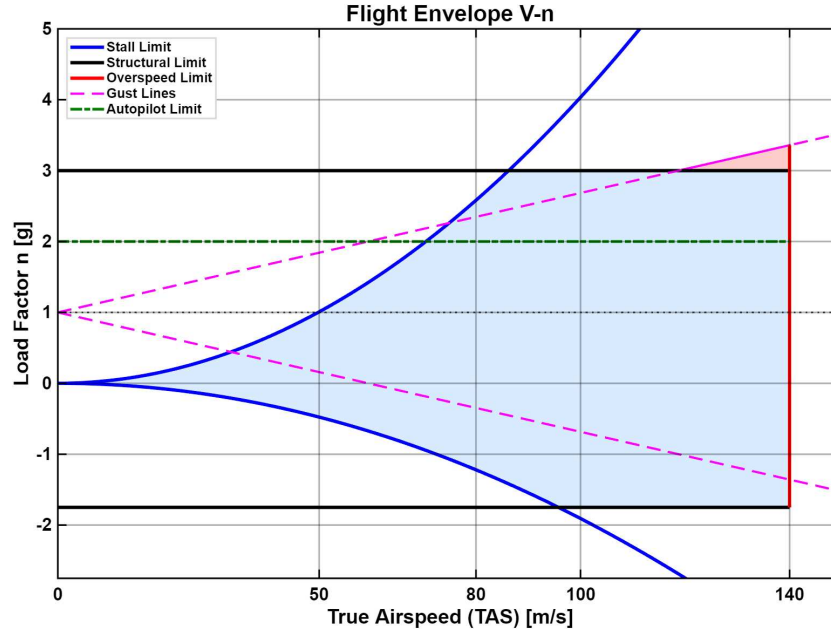


Figure 12.2: $V - n$ Flight Envelope.

12.3 Mission Constraints

The results provided by this analysis, highlight some important constraints to guarantee for the safety of the aircraft:

1. **Maneuvering Speed:** In the $V-n$ envelope, it can be seen that the intersection of the aerodynamic stall curve and the $3g$ limit occurs at approximately $V_A \approx 90 \text{ m/s}$. this implies that below this speed, the aircraft will stall before breaking; above this speed, full control deflection could cause structural failure.
2. **Bank Angle Restriction:** Since the autopilot is limited to $n = 2.0$, and considering that in a coordinated turn $n = 1/\cos(\mu)$, the maximum bank angle is confirmed to be limited ad $\mu = 60^\circ$, as was presented in Table 8.1.
3. **Turbulence Mode:** If, while cruising at high speed, the flight computer detects gust of wind approaching the 15 m/s limit, it must be programmed to automatically reduce the airspeed below the intersection of the gust line and the structural limit ($\approx 125 \text{ m/s}$) to prevent structural damage.

13 | Mission Simulation

Having defined all the main characteristics of the proposed design for this UAV, a simulation is now performed in more realistic conditions, to better visualize the actual, potential performance of the system. Therefore, this chapter will discuss a few simple flight missions in *Arcadia Planitia*, comparing these results with the reference ones, detailed in Chapter 8, which are referred to a flight performed at 1000 *m* of altitude above the ground, simplified as the datum; on Mars, however, areas with these topographic conditions are relatively rare.

Arcadia Planitia has been chosen as the staging theater for this simulation as it emerged as the most valid option among the four presented in Section 3.5: it is one of the most scientifically and operationally attractive regions on the planet. Moreover, its vast, flat lowlands offer a safe landing environment for the UAV, while the extensive presence of subsurface ice makes it a prime candidate for In-Situ Resource Utilization [1]. Furthermore, the region has gathered significant attention from the commercial space sector; the private aerospace company SpaceX has explicitly identified *Arcadia Planitia* as a primary target for its future *Starship* missions, citing the optimal balance between resource availability, crucial for propellant production, and low latitude [28]. Developing the simulation on such a high-priority exploration zone ensures that the mission profile presented in this thesis remains close to a potential real-world scenario.

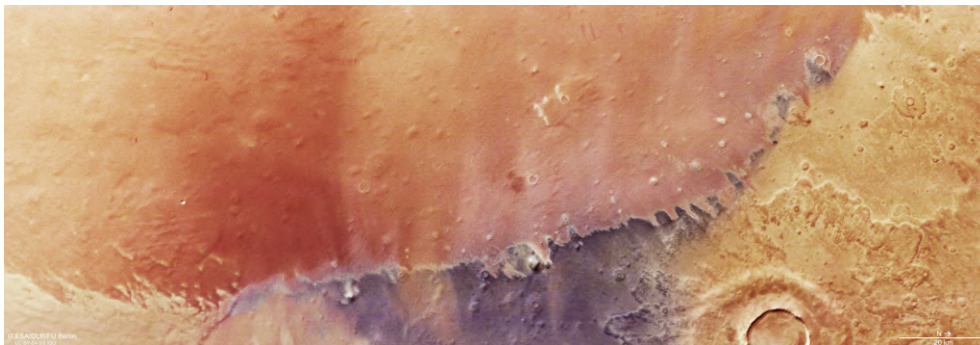


Figure 13.1: Satellite view of *Arcadia Planitia*

13.1 Local Conditions

To maximize the realism of the simulation, the local environmental parameters were calculated specifically for *Arcadia Planitia*. The crucial factor is the elevation: being a lowland, the atmospheric density in the region is higher compared to the reference datum. Using the exponential atmospheric model defined in Section 3.3, and assuming a local mean surface temperature slightly higher than the planetary average, the

operating conditions were derived as follows:

Quantity	Symbol	Value
Geographic Coordinates	LAT. LONG.	46.00°N, 185°E
Local Elevation	h_{local}	−3800 m
Local Gravity	g_{loc}	3.725 m/s ²
Mean Temperature	T_{loc}	215 K
Speed of Sound	a	232.3 m/s
Atmospheric Density	ρ_{loc}	0.0285 kg/m ³

Table 13.1: Local environmental parameters in *Arcadia Planitia*

These results show a 42.5% increase in air density which, as will be shown, will have an effect on the overall flight performance of the aircraft.

Being a vast plain, *Arcadia Planitia* is subject to peculiar wind patterns: during the day, slope winds tend to flow upwards from the depression towards the southern highlands, while the cycle reverses at night; crosswinds during the takeoff and landing phases will be an aspect to not underestimate. However, the flatness of the region minimizes mechanically induced turbulence caused by topographic obstacles [1].

13.2 Mission Profile

In this section, a complete and detailed mission profile is analyzed. The mission is assumed to launch from a hypothetical human settlement, located in *Arcadia Planitia* at the coordinates indicated in Table 13.1. Here, the UAV can be properly refueled and maintained by the astronauts, while supporting their scientific activities.

This ground station would have two fundamental capabilities, which are essential for the flight operations of the aircraft:

- **Fuel Production:** Thanks to the abundance of subsurface water ice in the region, the human crew would be able to produce the required propellant (Liquid Methane) and oxidizer (LOX) to refuel the UAV. The production cycle relies on the Sabatier process and water electrolysis, technologies that are also critical for sustaining the base’s life support systems, as detailed in Section 7.3 [26].
- **Runway Availability:** Based on the preliminary calculations, the nominal takeoff run for the UAV+ETR system is estimated at approximately 300 m.

Consequently, the base site must feature a cleared landing strip of 450 m , considering a safety factor of 1.5 to account for performance variations.

Moreover, it is important to address the issues regarding a Rejected Takeoff (RTO), since neither the UAV nor the External Takeoff Rig are fitted with a conventional braking system; they rely solely on the friction between the aluminum skid and the ground. Due to the reduced Martian gravity, the vertical load on the skid is low, resulting in limited braking efficiency, thus, in the case of a critical malfunction before takeoff speed, the system would likely overrun the designated strip and slide into the surrounding regolith. However, the ETR's propulsion system, based on the *VR900* engine, was selected specifically for its high reliability and simplicity, minimizing the probability of a sudden thrust termination during the initial seconds of the launch [37].

13.3 Takeoff Phase

The same numerical simulation which was presented in Chapter 9 was used, this time considering the local conditions reported in Table 13.1; results were encouraging, and further confirmed the choice of this region of Mars as being suitable for this kind of mission.

In particular, the simulation reveals a crucial advantage: the increased atmospheric density drastically lowers the required takeoff speed, as well as it reduces the take-off run. The aircraft generates sufficient lift to separate from the ground at just 49.2 m/s , compared to the $\approx 76\text{ m/s}$ required at the datum altitude; thus, the resulting ground roll is just 217 m . This result validates the safety of the proposed 450 m strip, as the aircraft uses only 48% of the available length to take off.

Parameter	Value
Takeoff Speed (V_{TO})	49.16 m/s
Takeoff Run Distance (d_{TO})	216.53 m
Time to take off (t_{TO})	8.81 s
Peak Thrust Used	2104 N ($\approx 68\% T_{MAX}$)
Average Net Acceleration	5.58 m/s^2

Table 13.2: Simulated Takeoff Performance in *Arcadia Planitia*

An additional, interesting finding from the simulation is the thrust requirement; it can be noted that to achieve the required performance, the engine does not need to run at its maximum rating of 3100 N ; the simulation indicates that a thrust setting of roughly 2100 N is sufficient [37].

A graphical visualization of this phase can be seen in Figure 13.2:

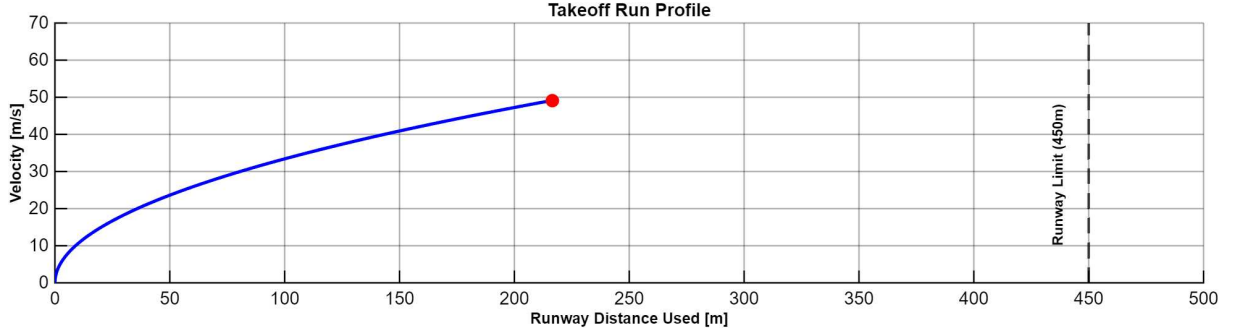


Figure 13.2: V_{TO} against d_{TO}

13.4 Cruise Phase

At the end of Phase 3 of the takeoff sequence (Section 9.1), the UAV is detached from the ETR, and is now free to start its cruise at a height of 1000 m above the ground; thus, the altitude is now -2800 m . As per Section 8.2, the simulation analyzed three distinct aerodynamic configurations to identify the optimal flight parameters, each one of them using four different cruise strategies; however, since previous results highlighted the constant attitude cruise program as the most suited for the mission requirements, it was the only one indicated in Table 13.3, for each attitude configuration.

Parameter	Conf. A	Conf. B	Conf. C
Angle of Attack (α)	1°	3°	0.88°
Lift Coefficient (C_L)	0.36	0.60	0.35
Aerodynamic Efficiency (E)	33.6	39.5	32.9
True Airspeed (Start)	103.0 m/s	79.8 m/s	105.1 m/s
Max Endurance	3.98 h	4.67 h	3.89 h
Max Range	1302 km	1186 km	1300 km

Table 13.3: Cruise Performance Comparison at Constant C_L

As highlighted by the data, Configuration B exhibits superior performance in terms of endurance, offering nearly 4.7 hours of flight time, approximately 18% higher than Configuration A. However, as established in Subsection 8.4.3, the primary operational requirement is to maximize the surface area mapped during a single

flight, which is directly proportional to the total distance covered; for this reason, Configuration A is selected as the primary cruise strategy, thanks to its superior range. Nevertheless, the flight control architecture remains flexible, allowing the system to adopt alternative configurations should specific mission scenarios require it.

To ensure consistent data quality for the optical payload, the flight control computer is programmed to execute a terrain following trajectory, rather than a simple constant altitude cruise. By combining pre-loaded Digital Terrain Models (DTM) with inputs from the altimeter, the autopilot can continuously adjust the pitch attitude to maintain a target vertical clearance of 1000 m above ground level (AGL). It is reasonable to expect that, however, this terrain following strategy, despite having relevant advantages in terms of the quality of data gathered, will affect the overall cruise performance; in particular, a reduction in actual cruise range is to be expected, due to the continuous climbs and descents that the UAV would perform in order to maintain a constant height. This altitude variation was accounted for in the numerical simulation; thanks to *Arcadia Planitia* being relatively flat, the maximum deviation from the nominal flight path was observed to be $\pm 113\text{ m}$, with average values being as low as $\pm 35\text{ m}$, depending on the route chosen. Nevertheless, this main factor and the possibility of thrust variations due to other external factors, such as wind, were accounted for, and the maximum range available for the mission was conservatively set at 1150 km .

13.5 Digital Terrain Model Generation

To better comprehend and visualize it, the terrain following logic was not implemented on a flat surface, but on a real topographic dataset of the target region. The elevation data was retrieved from the Mission Experiment Gridded Data Record (MEGDR) generated by the Mars Orbiter Laser Altimeter (MOLA) instrument, which flew aboard the NASA Mars Global Surveyor spacecraft [29].

For this simulation, a specific subset corresponding to the coordinates of the area of interest was extracted and processed in MATLAB; the resulting Digital Terrain Model (DTM) represents an accurate 3D representation of the *Arcadia Planitia* surface. This 3D model then served as the staging ground for the simulation: at each time step of the flight path, the script calculates the exact ground elevation and subsequently computes the required aircraft altitude as $z_{flight} = z_{ground} + 1000\text{ m}$ to maintain the target height. It should be noted that, given the vast distances traveled, in all of the following graphical representations vertical distances have been enhanced.

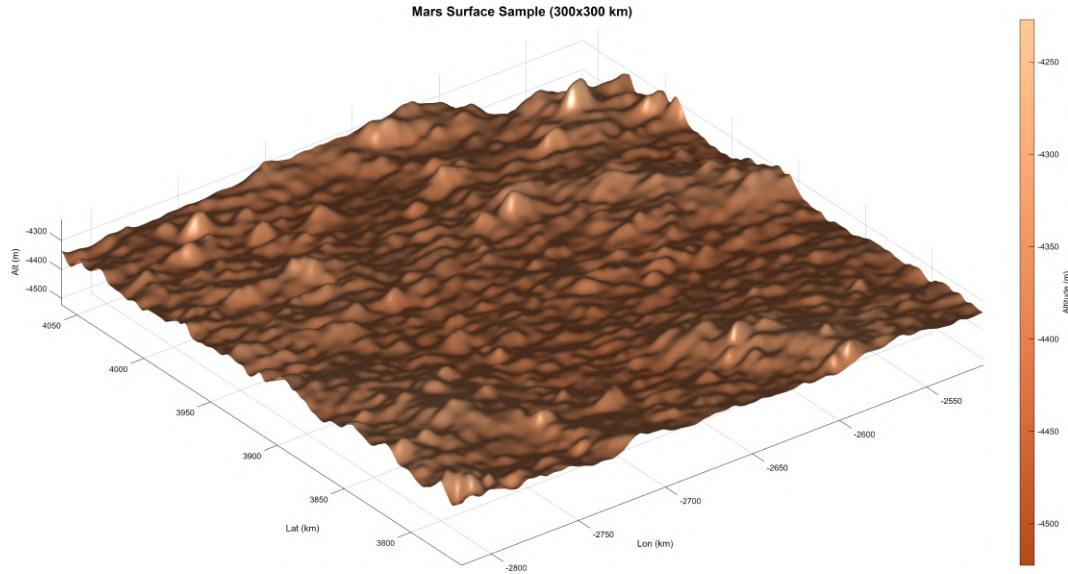


Figure 13.3: 3D model of a portion of *Arcadia Planitia* generated on MATLAB

13.6 Possible Flight Paths

To properly visualize the flexibility of the UAV, a few flight path geometries were defined and simulated. Before detailing the trajectories, it is essential to distinguish between the two vertical reference frames adopted in the cruise logic, as they fundamentally alter the behavior of the aircraft relative to the terrain:

- **AGL (Above Ground Level):** This reference frame measures the altitude of the UAV directly relative to the terrain surface vertically beneath it. Maintaining a constant AGL corresponds to a terrain following strategy, in which the aircraft actively climbs and descends to match the topographic profile.
- **AMSL (Above Mean Sea Level / Datum):** This refers to the altitude relative to the planetary reference datum; maintaining a constant altitude in this frame is simpler for the autopilot, but results in a varying distance from the ground, potentially compromising data consistency or safety over uneven terrain.

In the following simulations, graphic results are presented both for AGL and AMSL strategies.

13.6.1 S-Shape Trajectory

The first simulated path, designated as an S-shape, represents an open-loop exploration flight, which does not conclude on the starting location. This flight path is characterized by a sinusoidal ground track that allows the UAV to cover a wide

lateral area while progressing towards a distant target, presumably a second settlement, and it allows to take full advantage of the long-range capabilities of the drone.

First of all, a 2D representation of the trajectory is shown, to fully visualize the difference between the AGL and AMSL strategies. While on 3D graphics the dif-

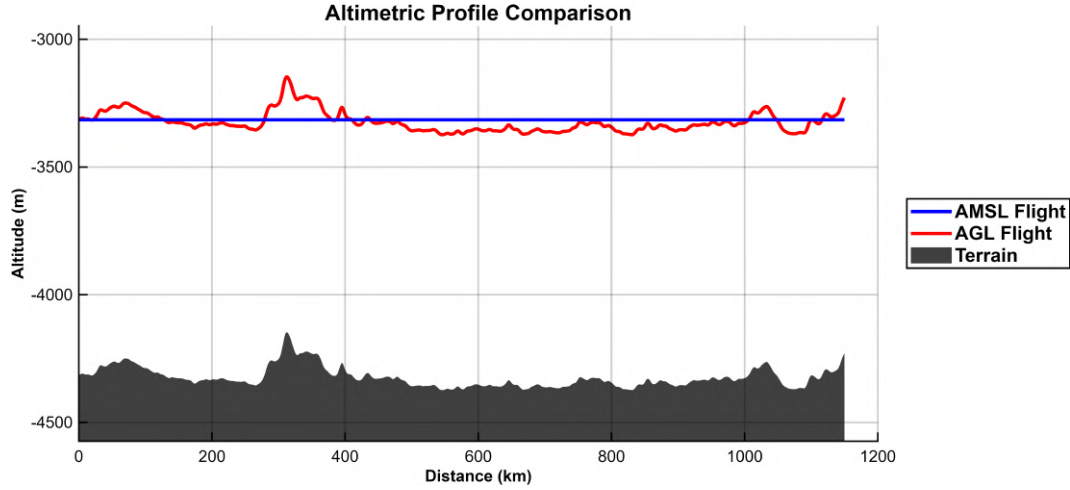


Figure 13.4: 2D profile for the S-shape trajectory

ference in altitude variations during the cruise is less notable, they still provide an interesting view of how the trajectory plays out on the Martian environment.

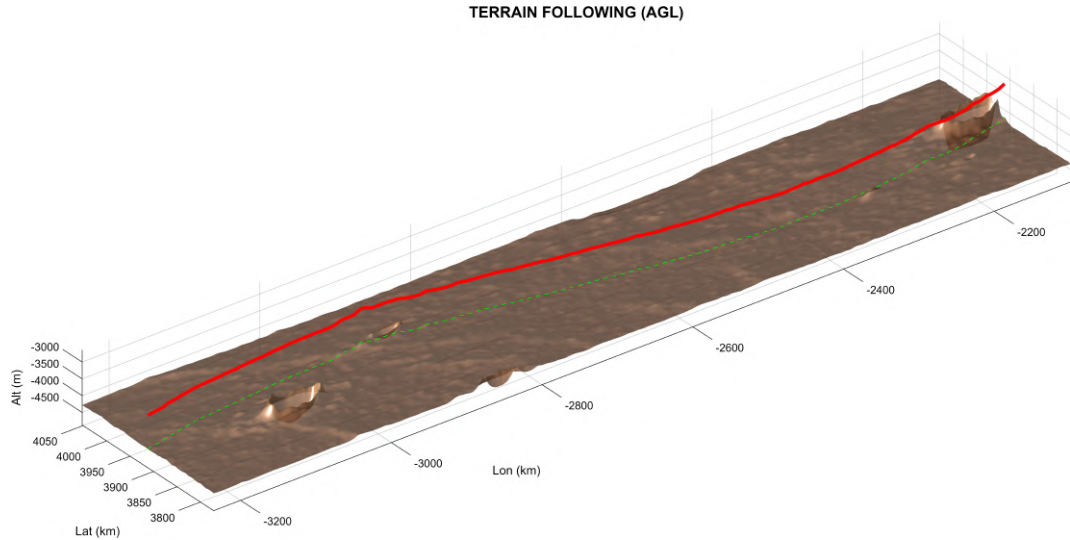


Figure 13.5: AGL S-shape flight path

13.6.2 Closed Loop Trajectory

The second trajectory defines a closed-loop oval pattern. Unlike the S-Shape, which is designed to cover the maximum distance possible, this profile is intended for continuous loitering over a specific area, featuring a coordinated 180° turn to return

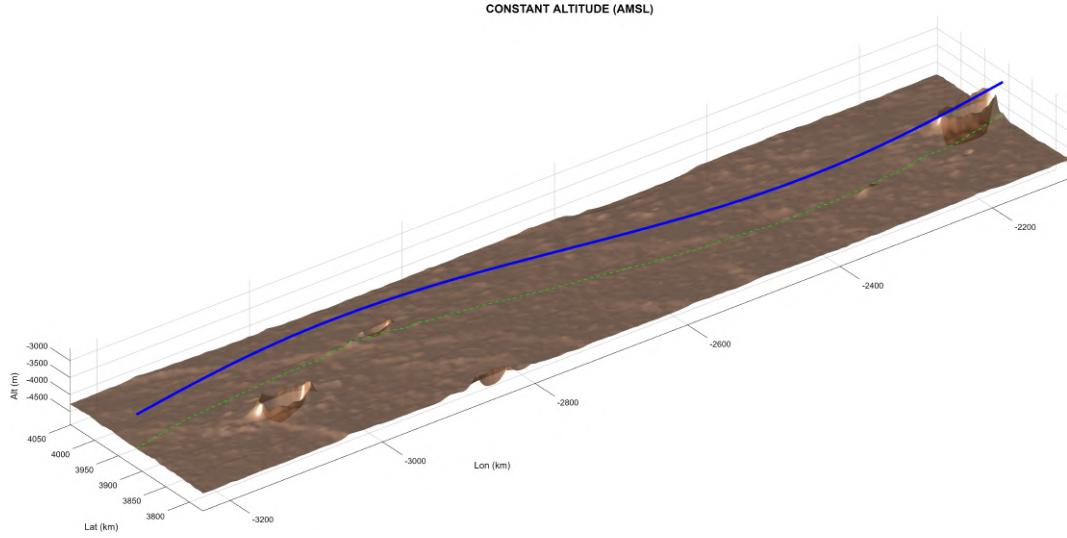


Figure 13.6: AMSL S-shape flight path

back to the starting position. Since the area of operation is the same as for the S-Shape trajectory, the 2D visualization is omitted, and only the AGL profile is included, to show how the flight path appears.

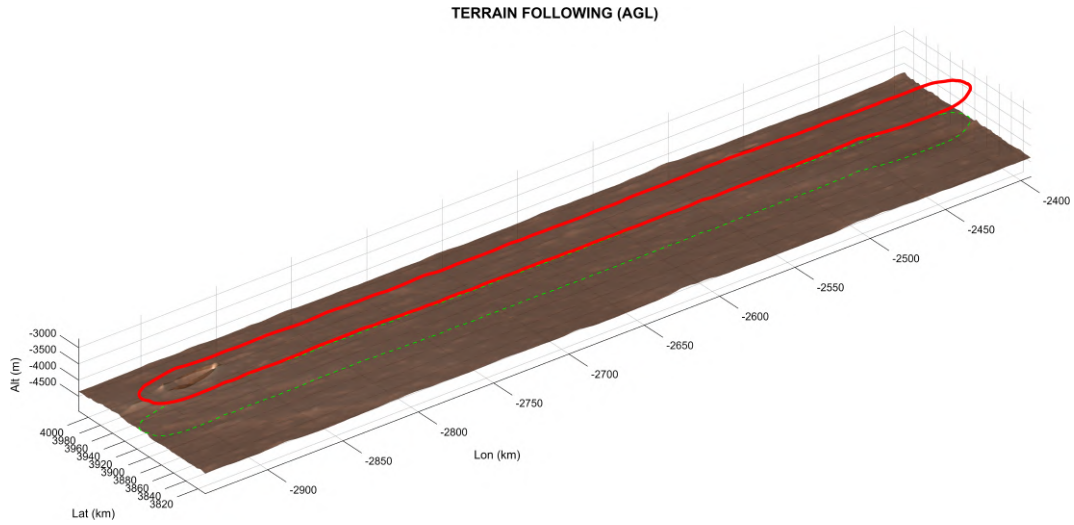


Figure 13.7: AGL closed-loop flight path

13.6.3 Examples of Other Possible Trajectories

Thanks to the great range and endurance capabilities of the UAV, the possible trajectories are virtually unlimited, depending on the scenario and the mission required. Just as an example, two possible search patterns have been simulated, given their high potential utility in the early stages of the mission. Despite staying in the close proximity of the runway, these patterns allow for an excellent overview of the immediate surrounding and the possibility to create an even more accurate map of the

area in which the astronauts will be physically present.

The two patterns proposed are:

- **Spiral Trajectory:** Once the drone is released at cruise altitude, it follows an inside-out circular trajectory, progressively widening the radius of turn. Once the range has been depleted, it will land back to the starting position.
- **Flower-like Trajectory:** It allows for a complete overview around a central point, in which the four "leaves" converge. Its applications are very similar to the ones of the Spiral.

Only the 3D AGL trajectories are presented for these two last flight paths.

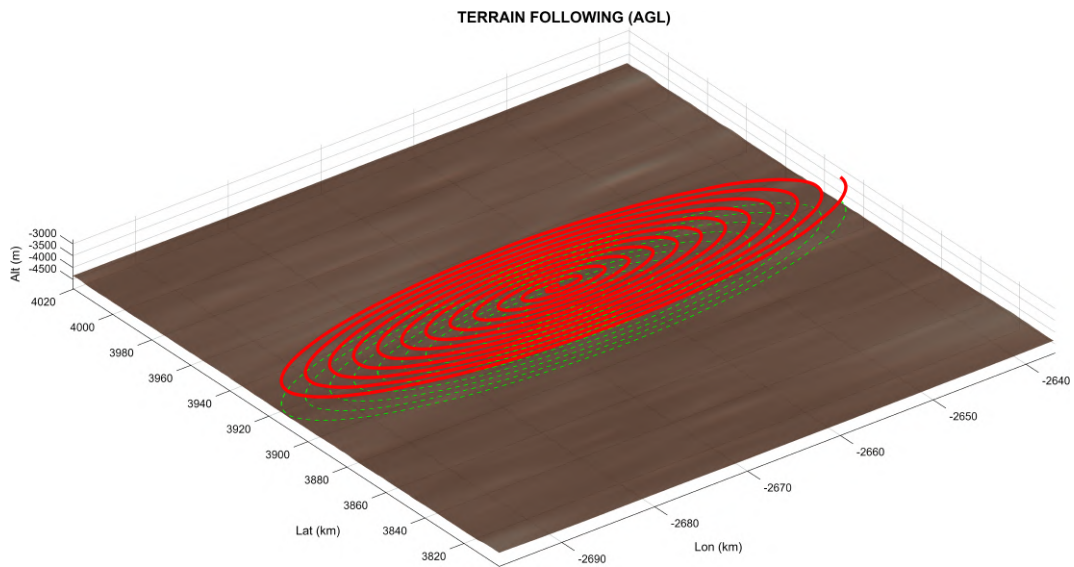


Figure 13.8: AGL Spiral flight path

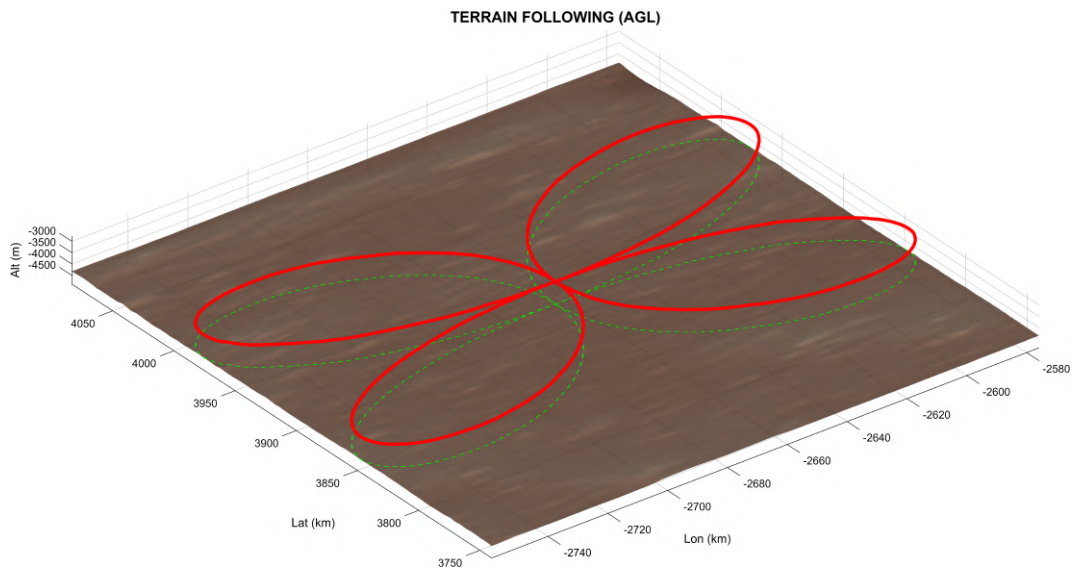


Figure 13.9: AGL Flower-like flight path

14 | Conclusions

The exploration of Mars is continuously evolving, transitioning towards a future in which crewed missions to the Red Planet will become a reality. In this scenario, the focus shifts from pure scientific exploration to a more complex operational environment, in which safety and resource utilization must be prioritized. The work presented in this thesis aimed to bridge the gap between orbital satellite observation and the slow process of rover exploration, proposing an unmanned aerial platform capable of acting as a long-range scout for future astronauts in *Arcadia Planitia*.

The feasibility analysis and simulations conducted showed encouraging results, with numerical data confirming that a reusable, rocket-propelled UAV represents a viable solution for the Martian environment; moreover, the design fulfills all the identified mission requirements. A fundamental aspect of this work lies in the design philosophy adopted. While the mission scenario is completely hypothetical, the engineering approach followed throughout the project was kept as close to reality as possible; in fact, the development of this UAV did not rely on ideal assumptions or basic estimations but followed a structured, iterative engineering path, guided by the multidisciplinary knowledge acquired during this Master's Degree program.

14.1 Operational Impact

Beyond the raw numbers, the true value of this system lies in its simplicity and reliability, which make it a dependable asset for future crewed missions; for a crew operating on an alien surface, time and uncertainty will be the greatest threats. This UAV serves as the "eyes" of the expedition: whether executing a linear S-Shape exploration to chart a safe path for a rover, or performing a loiter to inspect a potential site for habitation, the drone provides the crew with high-resolution, 3D situational awareness that orbiters simply cannot match, and that traditional surface missions would take years to produce.

14.2 Future Developments

While this thesis establishes a solid conceptual foundation, supported by extensive flight dynamics research, the path to a final design surely requires further investigation. A few aspects future work should focus on are:

- **Propulsion System Testing:** Static fire tests of the rocket nozzle geometry in a vacuum chamber to validate the theoretical results and consumption models developed in the simulations.

- **Autonomous Landing Logic:** While the takeoff and cruise phases were the primary focus of this study, landing on a prepared strip remains a critical phase requiring dedicated control laws and potentially a drag-chute system for deceleration.
- **Aerodynamic and Experimental Validation:** While the numerical results presented in this study provide a solid baseline, they are inherently limited by the low-fidelity nature of the preliminary design tools. Future development phases must include high-fidelity CFD analysis and wind tunnel testing to refine the aerodynamic coefficients and stability derivatives. Furthermore, a validation campaign could involve a stratospheric flight test: releasing a scale model from a high-altitude balloon at approximately 30 km in Earth's atmosphere would effectively simulate the Reynolds number and dynamic pressure conditions encountered on Mars, providing crucial data to validate the numerical models.

Ultimately, this thesis demonstrates that the proposed vehicle is not merely a theoretical exercise, but a viable engineering baseline for future development, proving that the Martian atmosphere is not a barrier, but a new possibility for human exploration.

XFLR5 Experimental Results

This Appendix contains the complete results obtained during the XFLR5 simulations, both for the clean-wing configuration and two different flap configurations.

α [°]	C_L	C_D	C_m	E (L/D)
-3.0	0.0586	0.0092	0.3730	6.39
-2.5	0.1190	0.0093	0.3420	12.75
-2.0	0.1793	0.0096	0.3102	18.72
-1.5	0.2397	0.0100	0.2777	24.06
-1.0	0.3000	0.0105	0.2446	28.60
-0.5	0.3603	0.0112	0.2108	32.25
0.0	0.4205	0.0120	0.1763	35.07
0.5	0.4807	0.0130	0.1412	37.03
1.0	0.5409	0.0141	0.1054	38.30
1.5	0.6009	0.0154	0.0691	38.95
2.0	0.6610	0.0169	0.0313	39.09
2.5	0.7209	0.0186	-0.0054	38.82
3.0	0.7807	0.0204	-0.0435	38.31
3.5	0.8405	0.0224	-0.0822	37.59
4.0	0.9002	0.0245	-0.1214	36.71
4.5	0.9597	0.0269	-0.1611	35.74
5.0	1.0191	0.0293	-0.2015	34.75
5.5	1.0784	0.0320	-0.2423	33.74
6.0	1.1376	0.0347	-0.2836	32.75
6.5	1.1966	0.0377	-0.3254	31.75
7.0	1.2555	0.0409	-0.3677	30.70
7.5	1.3242	0.0443	-0.4104	29.92
8.0	1.3727	0.0478	-0.4535	28.70
8.5	1.4311	0.0516	-0.4971	27.72
9.0	1.4893	0.0558	-0.5410	26.71
9.5	1.5473	0.0604	-0.5851	25.61
10.0	1.6051	0.0657	-0.6294	24.44
10.5	1.6627	0.0710	-0.6741	23.41

Table 1: Aerodynamic Polar Data - Flap 5° Configuration

α [°]	C_L	C_D	C_m	E (L/D)
-1.0	0.1208	0.0086	0.1422	13.98
-0.5	0.1811	0.0090	0.1091	20.19
0.0	0.2415	0.0094	0.0754	25.66
0.5	0.3018	0.0100	0.0410	30.15
1.0	0.3621	0.0108	0.0060	33.65
1.5	0.4224	0.0117	-0.0297	36.25
2.0	0.4826	0.0127	-0.0659	38.00
2.5	0.5427	0.0139	-0.1027	39.04
3.0	0.6028	0.0153	-0.1401	39.53
3.5	0.6628	0.0168	-0.1780	39.52
4.0	0.7227	0.0185	-0.2165	39.15
4.5	0.7825	0.0203	-0.2555	38.55
5.0	0.8423	0.0223	-0.2951	37.77
5.5	0.9019	0.0244	-0.3352	36.92
6.0	0.9614	0.0267	-0.3758	36.01
6.5	1.0208	0.0291	-0.4169	35.03
7.0	1.0800	0.0318	-0.4585	33.99
7.5	1.1391	0.0346	-0.5004	32.94
8.0	1.1981	0.0375	-0.5428	31.91
8.5	1.2569	0.0407	-0.5857	30.92
9.0	1.3155	0.0439	-0.6290	29.96
9.5	1.3740	0.0473	-0.6727	29.04
10.0	1.4323	0.0509	-0.7168	28.15
10.5	1.4904	0.0546	-0.7613	27.31
11.0	1.5483	0.0584	-0.8062	26.50
11.5	1.6060	0.0624	-0.8515	25.74
12.0	1.6635	0.0665	-0.8971	25.02
12.5	1.7208	0.0707	-0.9430	24.33
13.0	1.7779	0.0751	-0.9893	23.67
13.5	1.8347	0.0797	-1.0358	23.03
14.0	1.8913	0.0844	-1.0826	22.42
14.5	1.9477	0.0893	-1.1300	21.82
15.0	2.0037	0.0944	-1.1770	21.23
15.5	2.0596	0.0999	-1.2244	20.62
16.0	2.1152	0.1059	-1.2718	19.97

Table 2: Aerodynamic Polar Data - No Flap Configuration

α [°]	C_L	C_D	C_m	E (L/D)
3.5	1.1231	0.0336	0.1113	33.45
4.0	1.1823	0.0365	0.0710	32.43
4.5	1.2413	0.0395	0.0300	31.46
5.0	1.3001	0.0426	-0.0115	30.53
5.5	1.3588	0.0458	-0.0536	29.65
6.0	1.4173	0.0492	-0.0926	28.80
6.5	1.4757	0.0528	-0.1394	27.97
7.0	1.5339	0.0565	-0.1830	27.17
7.5	1.5918	0.0600	-0.2272	26.53
8.0	1.6496	0.0643	-0.2718	25.64
8.5	1.7072	0.0686	-0.3169	24.90
9.0	1.7646	0.0730	-0.3624	24.19
9.5	1.8218	0.0776	-0.4082	23.47
10.0	1.8787	0.0825	-0.4545	22.78
10.5	1.9354	0.0877	-0.5011	22.08
11.0	1.9919	0.0932	-0.5480	21.38
11.5	2.0481	0.0993	-0.5951	20.63
12.0	2.1051	0.1061	-0.6423	19.85
12.5	2.1598	0.1160	-0.6883	18.62

Table 3: Aerodynamic Polar Data - Flap 10° Configuration

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