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A feasibility study of a future European Single-Stage-To-Orbit (SSTO) spaceplane

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ABSTRACT

This paper synthesizes the development and feasibility system studies of a horizontal take-off and landing Single Stage To Orbit (SSTO). This medium-sized spaceplane is planned to perform manned flights with 4 crew members or up to 3 tons of payload into Low Earth Orbit (LEO). Thanks to its horizontal take-off and landing, it will be able to operate from any conventional runway.

The results of the design iterations for such an aircraft will be presented, considering various sub-studies conducted to inform design choices. This includes the logic and outcomes of the aircraft's airframe design, encompassing its external structure, aerodynamic performance, internal structure, layout, and methods for accurate weight estimation. The aircraft being planned to use airbreathing propulsion technologies and having to execute atmospheric re-entries, the trajectory of a mission, its optimization, the aerothermal constraints, and the associated thermal management strategy are taken into consideration. The efficiency of the propulsion system employed plays a critical role in achieving low Earth orbit while maintaining a payload. Thus, from the state of the art and on proposal of innovative concepts, a propulsion system is proposed as well as an estimation of its performances. In order for manned flights to be conducted, they must adhere to multiple safety standards, and it is necessary to characterize the aircraft's reusability. Strategies to address these criteria will be developed based on the analysis of risks, reliability, and operational safety.

Finally, the designed spacecraft being proposed for the market, it is necessary to approach the place that such a device would have and how such a project could be conducted. We will synthesize the study of the current market, the estimate of the costs of design, implementation, associated maintenance and finally the financial sustainability of such a project.

NOMENCLATURE

SSTO	Single Stage To Orbit
LEO	Low Earth Orbit
GSC	Guiana Space Center
LOX	Liquid Oxygen
LH2	Liquid Hydrogen
RP-1	Refined kerozen
CH4	Methan
C3H8	Propane
Cl	Lift Coefficient
Cd	Drag Coefficient

INTRODUCTION

The past two decades have witnessed the emergence of a new ecosystem comprised of companies and organizations dedicated to pushing the boundaries of space activities and capabilities. With the continuous reduction in costs and increased accessibility to space, we are entering a future where limitless possibilities await. As we witness the advancements in reusability, recycling, and the development of new propulsion systems for space launchers, it is natural to contemplate what milestones lie ahead in this field. Throughout history, numerous projects have aimed to replace the conventional staged launchers, seeking to enhance reusability and operational efficiency. Some of these projects are nearing completion, while others are still in the development phase, adapting to the availability of necessary technologies. One certainty remains: an SSTO vehicle will eventually take flight.

Embracing this challenge, the AndroMach team has embarked on a daring endeavor to design a revolutionary single-stage space vehicle. The ultimate objective is to redefine access to space by presenting a viable alternative to traditional launch systems. During the pre-project phase, extensive studies, iterative design processes, and the application of cutting-edge technologies have led to the development of an innovative SSTO concept. The team has been mindful of the latest advancements in reusability, propulsion, as well as the aerodynamic, aerothermal, recycling and safety considerations inherent in manned flight.

With the backdrop of technological progress, it prompts us to question the feasibility of such an aircraft. What solutions does AndroMach propose? And what role would this vehicle play within the future space ecosystem?

Development logic

The pre-project study is conducted in an iterative manner. The initial version of the aircraft, referred to as VH0, was developed to meet the initial specifications derived from an adaptation of the Hyperion project [1]. Through the analysis of VH0 and the findings from the studies, a subsequent iteration of the aircraft, VH1, was designed. VH1 provided valuable insights that facilitated the refinement of the concept through operational decision support, leading to the most recent iteration (Figure 1). This discussion will focus on the VH2 iteration, which introduces an alternative option, VP0, utilizing C3H8. Further investigations are still underway and will be the subject of future publications to refine the project's contours.

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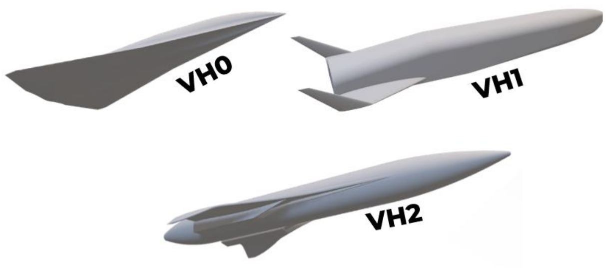

Figure 1: *Different vehicle versions studied in the process*

In order to highlight the interactions between each task group, the utilization of a Design Matrix Structure (DMS) has been implemented to establish a comprehensive framework for the project. This structure was developed following a thorough literature review to identify pertinent areas of study and establish connections between them. The diagram below shows the different task group and their connections (Figure 2).

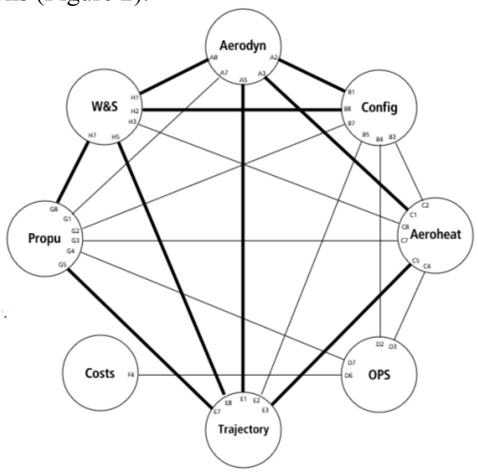

Figure 2: *The DMS of the project*

Aerodyn: Aerodynamic, **Config:** Internal configuration, **Aeroheat:** Aeroheating, **OPS:** Operations, **Trajectory:** Trajectory, **Cost:** Cost, **Propu:** Propulsion, **W&S:** Weight & Structure.

Operational Decision Support (ODS)

The methodology employed for Operational Decision Support (ODS) is ELECTRE I. Its aim is to assign scores to each solution based on a set of criteria. These scores can be weighted to reflect the relative importance of each criterion. Concordance and discordance matrices are then utilized to generate an outranking matrix, which identifies the optimal solution. It is crucial to emphasize the significance of expert opinions during the rating process. Based on the results of VH0 and VH1, 4 concepts are proposed (Figure 3). The first, Concept A, is a version similar to VH1, used here as a reference concept. Concept B is a version similar to the SSTO Skylon launcher. Concept C, on the other hand, is more exotic, since it has the particularity of making an atmospheric re-entry on its back, i.e. in the opposite direction to the ascent phase. Concept D is a variant of C, where only the air breathing phase is on the back. Take-off, landing and re-entry are performed with the engines on top of the wings.

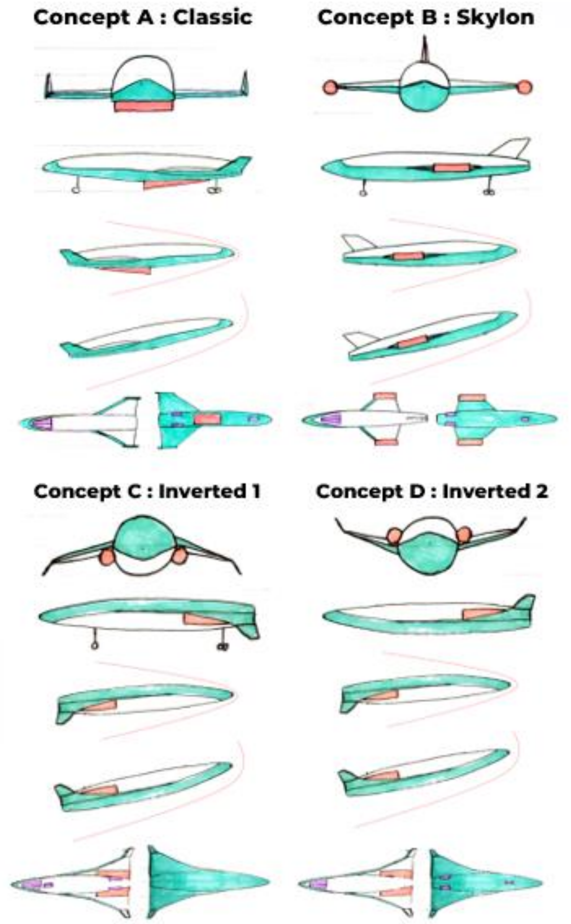

Figure 3: *The 4 concepts studied in ODS, heat shield (green), engines and air intake (orange), doors (purple)*

The criteria defined and their direct impact are listed in Table 1 below. The results of the study place Concept C well ahead, followed by Concept D and B, which are fairly close. Concept A comes last. The Concept C stood out for various reasons. Firstly, it excels in optimizing the aerodynamic shape necessary for flight and provides effective protection to the propulsion system during atmospheric re-entry. Additionally, it streamlines the development process by eliminating the necessity for heat-shield doors. The feasibility of the two roll maneuvers is evident from the fact that the tanks are empty during this period. Due to its numerous advantages and other notable factors, the Concept C emerged as the preferred choice for VH2.

Impacts		Fonctions
PROPU	C1	Maximize air intake efficiency
PROPU	C2	Improve air intake homogeneity
RENTRY MASS AERODYN RELIAB	C3	Maximizing re-entry maneuverability
AERODYN	C4	Minimize aerodynamic drag
AERODYN	C5	Maximize the lift generated
RENTRY RELIAB	C6	Optimizing re-entry stability
AERODYN MASS	C7	Minimize moment generated by air inlets
MASSE RELIAB STRUCT	C8	Reduce heat shield discontinuities
AERODYN MASS STRUCT	C9	Minimiser la masse des ailes
OPS RELIAB	C10	Facilitating aircraft flight operations
STRUCT	C11	Facilitate propulsion subsystems placement
MASS STRUCT	C12	Minimizing landing gear size
PROPU RENTRY MASS OPS	C13	Minimize air intake thermal management
RENTRY OPS RELIAB	C14	Facilitate visibility on landing/take-off
OPS	C15	Comfortable for manned flights

Table 1: *Criteria used for ODS*

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THE SPACECRAFT CONCEPT

Concept Overview

The proposed spacecraft is a modestly sized vehicle, comparable in size to a launcher such as Vega C (Table 2). In the design iteration proposed here (VH2), the special feature is a "double-sided" vehicle, a concept evaluated in the ODS and developed on the following page. In accordance with the targeted dimensions, the aircraft is designed to accommodate 4 passengers. By excluding the survival equipment required for manned flight, this configuration must allow for a payload capacity of 3 tons in low Earth orbit (at least 400km, all inclinations). Table 2 below lays the foundations for a typical configuration, although this will have to evolve over time.

Table 2 :Characteristics of the spacecraft

Take-off	Horizontal	**Length**	28 m
Landing	Horizontal	**Span**	10 m
Propellants	LH2/LOX (or C3H8/LOX)	**Fuselage diameter**	2,6 m
Passengers	4	**Dry weight**	11 000 kg
Payload (LEO)	3 000 kg (400 km)	**Gross weight**	100 000 kg
Reusability	99,5%	**Air breathing propulsion**	2x Dual Mode Scramjet 300 kN
Max air breathing speed	Mach 10	**Rocket/ejector mode propulsion**	2x Rocket engines 200 kN

The spacecraft will therefore be available in two versions: the Crew version for manned flights and the Cargo version for payload launches. It is the Cargo version that will be equipped with doors for payload deployment.

Figure 4: *Representation of the spacecraft*

This concept has a special feature: its heat shield is on its "back" (Figure 5). This concerns an innovative concept, never attempted before. The idea is to perform the atmospheric re-entry on the back. In practice, the aircraft takes off while being belly-up, with a negative dihedral.

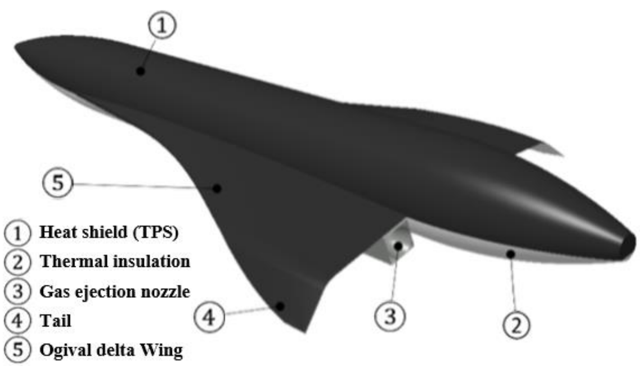

Figure 5: *View of the external structure of the spacecraft*

This configuration provides improved lift, allowing the entire ascent to be carried out in this position. Once in orbit, the aircraft performs a roll maneuver to position itself on its back, with a positive dihedral. This improves stability, and the atmospheric reentry is then performed in this position before conducting another roll maneuver shortly before landing. Thus, this concept allows for optimizing lift and stability when the aircraft requires it, as well as avoiding the need to place hatches in the thermal shield, which can cause sealing issues.

Mission Profile

The spacecraft is an SSTO based on an innovative mission profile which is described in Figure 6.

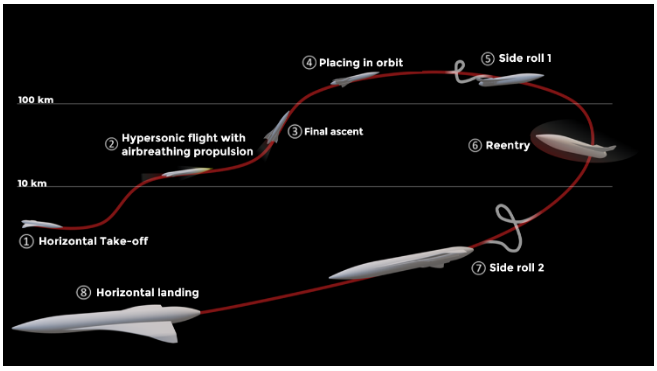

Figure 6: *Illustration of a typical mission*

(1) One of the spacecraft's strengths is its ability to take off from a conventional runway. As the dual-mode scramjet cannot be utilized before Mach 3, it is necessary to rely on either the turbojet or rocket engine for takeoff and reaching this speed. Previous research has indicated that the aircraft's weight exceeds the capability of taking off solely with turbojets, prompting exploration of alternative options like a an ejector ramjet mode using the two rocket engines.
(2) After igniting the ramjet propulsion mode at Mach 3, the aircraft transitions into the air breathing propulsion phase. Between Mach 5 and 6, the propulsion mode shifts to a scramjet with a supersonic combustion chamber, propelling the aircraft to Mach 10 (depending of used fuel). This phase occurs within the altitude range of 14 to 35 km.
(3) Once the air breathing propulsion reasonable limit has been reached, the propulsion switches to conventional rocket engine mode. This last mode gives the spacecraft its final injection speed into orbit and apogee needed.
(4) After reaching orbit, the vehicle is capable of fulfilling its assigned missions. The spacecraft utilizes its RCS (reaction control system) to execute the required orbital maneuvers. The duration can range from a few hours to several months.
(5) In preparation for atmospheric re-entry, the spacecraft employs its RCS to perform a roll maneuver, ensuring proper positioning of the heat shield. To optimize aerodynamics and aerothermodynamics, the heat shield has been positioned on the vehicle's back.
(6) Atmospheric re-entry then takes place without a hitch for a few tens of minutes. At the end of re-entry, the aircraft adjusts its trajectory and orientation to glide.

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(7) With the spacecraft back at subsonic speeds, the next step is to prepare for landing. To do this, the vehicle performs a second roll manoeuvre to reposition the landing gear.
(8) With the landing gear now on the correct side, the spacecraft glides to the planned landing site. During the approach phase, it opens its doors and deploys the landing gear. Once it touches down, the landing gear help it brake to a stop.

BASELINE DESIGN RESULTS

Aerodynamics

The external fuselage structure of the VH2 version has been designed as a compromise, aiming to balance the requirements for hypersonic flight (VH0) and atmospheric re-entry (VH1). In previous iterations, the configurations were focused on prioritizing one of these aspects over the other. The aircraft features a spherically blunted tangent nose, which provides a compromise between aerodynamic finesse and penetration for hypersonic flight, while ensuring optimal aerothermal performance. The central section of the aircraft, with a radius of 1.3m, follows the shape of the fuel tanks. The tapered tail design is of course designed to minimize drag.

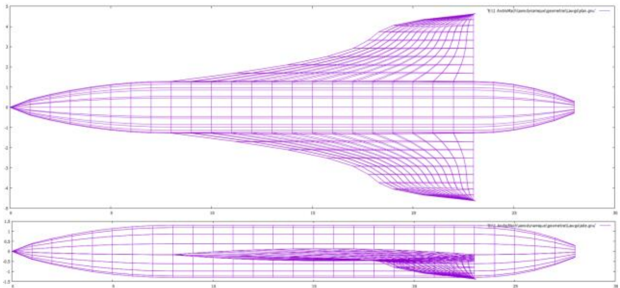

Figure 8: *Plan and Side view of the spacecraft in LWGS format*

The wing of the aircraft is an ogival delta wing, offering improved lift at low speeds by utilizing leading edge vortices and providing suitability for high speeds. With an aspect ratio of 1.36, the low aspect ratio allows for lift over a wider range of angles of attack, which is advantageous for our aircraft. The lenticular profile of the wing, with a rounded leading edge, is well-suited for hypersonic speeds, meeting our specifications. The maximum thickness of the wing is 8% of the chord, optimized for hypersonic glide ratio. The wing incorporates a dihedral of -7.5°, which provides stability and additional lift during hypersonic conditions. However, during atmospheric re-entry, when the aircraft is inverted, the negative dihedral becomes positive, offering stability during descent. The wing area is measured at 73m2 with an average chord length of 18.378m. The rudders are positioned at the wingtips, forming an angle of -38° with the wing, resulting in a 45.5° angle to the XY plane of the aircraft. This configuration reduces weight and overall height, which is beneficial during take-off and landing as it minimizes the downward pointing aspect.

To evaluate the geometry created in LWGS format (Figure 8), the PANAIR, following the user manual references of Code [2], and HAPB codes were employed to derive a matrix of lift and drag coefficients for the aircraft, considering the Mach number and the angle of incidence (Figure 9).

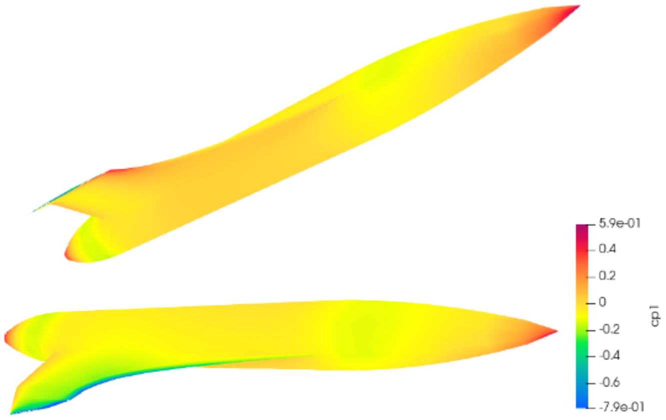

Figure 9: *Pressure coefficient on the fuselage according to PANAIR at Mach 0.5 and an angle of attack of 5°.*

The data was recalibrated using CFD software to incorporate viscous effects, as the Euler equations employed by PANAIR assume air to be non-viscous. Recalibrating the data was crucial in ensuring accuracy. The outcome was a correlation that effectively predicts the outcomes of the CFD studies. Additionally, this recalibration allowed for the validation of our PANAIR model, with the drag results exhibiting a relative error of less than 4%.

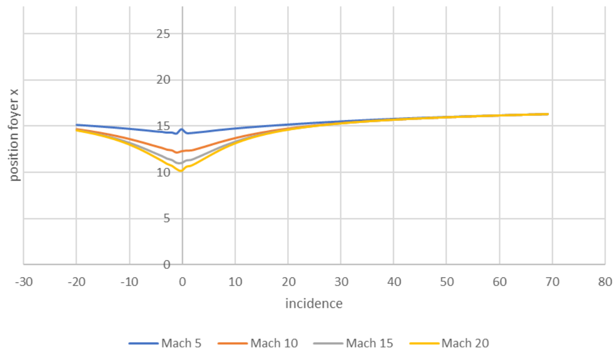

Figure 10: *Aerodynamic focal x position in function of Mach/Incidence*

Aerothermal Analysis

Without providing an exhaustive list of all types of thermal protection systems, it is clear that a reusable vehicle implies avoiding the use of ablative coatings. Therefore, it would be preferable to consider utilizing TPS (Thermal Protection System) tiles and active cooling for nose tip. However, since these tiles can be costly, the use of inexpensive ablative materials could be a good compromise for development prototypes. A cork coating, for instance, could prove to be highly effective, lightweight, easy to apply, and very cost-effective. Attempts have been made to assess the heat fluxes experienced during both the re-entry and ascent stages. However, the current findings do not provide sufficient data to establish a definitive thermal corridor. Since the aircraft functions as a load-bearing structure, the thermal stresses encountered during re-entry are expected to be lower compared to those experienced by a ballistic body.

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Internal Configuration

The internal structure of the aircraft follows a similar design to conventional launchers, consisting of thin-walled pressurized tanks. As depicted in Figure 11, liquid hydrogen (LH2) and liquid oxygen (LOX) tanks conform to the shape of the aircraft, sharing a common wall with the airframe. Due to the air-breathing propulsion system and the bulkiness of hydrogen compared to oxygen, the LH2 tank is larger than the LOX tank. Specifically, the LOX tank has a volume of 17 m3, while the LH2 tank has a volume of 71 m3. This results in carrying approximately five times more LOX than LH2.

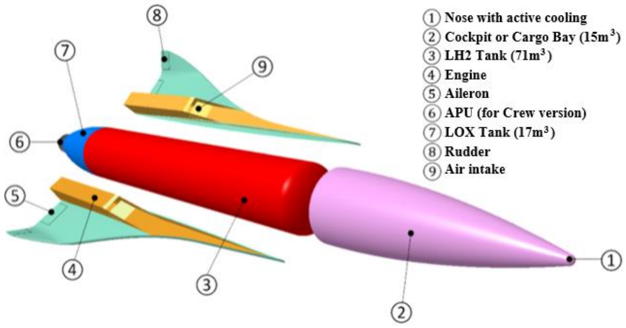

Figure 11: *Aircraft internal Layout*

The decision to use thin walls and integrate the airframe with the tank walls was made to reduce weight. Weight reduction and cost efficiency are key considerations in this version, aligned with a Design To Cost (DTC) approach. Located at the front of the aircraft is either a cockpit or a cargo bay, depending on the version. The cargo bay has dimensions of approximately 2.5 x 5 m and is designed to transport a payload of 3 tons into orbit. In the manned transport version, the cargo bay is replaced by a cockpit capable of accommodating four passengers. Weight distribution is a critical aspect of the project, necessitating meticulous calculations and organizational efforts to maintain the stability of the aircraft's center of mass during flight. Excessive fluctuations in the center of gravity, primarily attributed to propellant consumption, have the potential to affect the aircraft's performance. An example of mass balance is given in Figure 11 et 12.

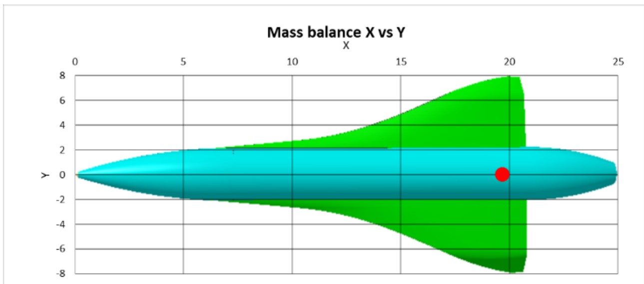

Figure 11: *Example of mass balance in XY plane*

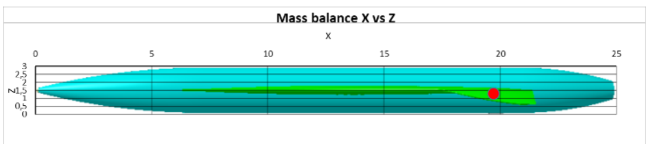

Figure 12: *Example of mass balance in ZX plane*

Mass Properties

A weight estimation tool has been developed in the form of a spreadsheet to facilitate the process of estimating the weight of the spacecraft. This tool incorporates a comprehensive database of over 200 equations that are used to estimate the weight of various sub-assemblies of the aircraft. The foundation of this tool is based on a document titled *Development of a Mass Estimating Relationship Database for Launch Vehicle Conceptual Design* by Reuben R. Rohrschneider [3]. This document serves as a valuable reference for mass estimation relationships, divided into different sections corresponding to specific sets whose mass can be estimated. The selection of this document as a reference is justified by two key factors. Firstly, it is relatively recent, published in 2002. Secondly, it derives its relationships from 12 different sources, thereby offering a broad range of models. This approach allows us to estimate the mass of the spacecraft with greater accuracy by selecting the most suitable models for each assembly being evaluated (e.g., certain models may be more appropriate for estimating the wing mass compared to the fuselage mass).To estimate the mass of the spacecraft a substantial set of input parameters needed to be provided. These parameters were primarily derived from the generated geometries of the aircraft. However, it is important to note that certain parameters were not solely based on geometric characteristics. These non-geometric parameters necessitated specific processing methods to ensure the most accurate and representative results possible. An example of previous mass budget is given in Table 3 with dry weight breakdown in Figure 13.

Table 3: *Example of previous mass budget of the spacecraft*

Items	Initial Weight (kg)	Last Weight (kg)
Wing & Tail Group	1 380	1 380
Body Group	3 754	3 754
Thermal Protection System	1 078	1 078
Main Propulsion	2 075	2 075
OMS/RCS Propulsion	250	250
Subsystems & others	2 500	2 500
Dry Weight	**11 037**	**11 037**
Payload to LEO	3 000	3 000
Other Inert Weights	220	220
Insertion Weight	**14 257**	**14 257**
LH2 Ascent Propellant	3 788	19 550
LOX Ascent Propellant	18 937	66 193
Gross Weight	**36 982**	**100 000**

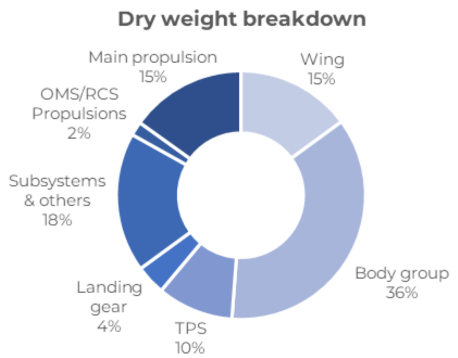

Figure 13: *spacecraft dry weight breakdown*

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Propulsion

The aircraft is equipped with two engines - one under each wing at the root - capable of operating in three different propulsion modes: ramjet, scramjet, and rocket engine (and ejector ramjet for take-off). This provides the aircraft with the ability to propel itself in both atmospheric conditions, utilizing ambient air, and in the vacuum of space. For the latest version of the aircraft, VH2, a flight plan consisting of three phases is being studied: rocket engine propulsion during takeoff to reach Mach 3 (ejector ramjet), utilization of the dual-mode scramjet up to Mach 10, and finally, completing the orbital insertion with the rocket engine. The configuration of this engine is depicted in Figure 14. The dual-mode scramjets are expected to provide a maximum thrust of 200 kN individually versus 300 kN for rocket engines.

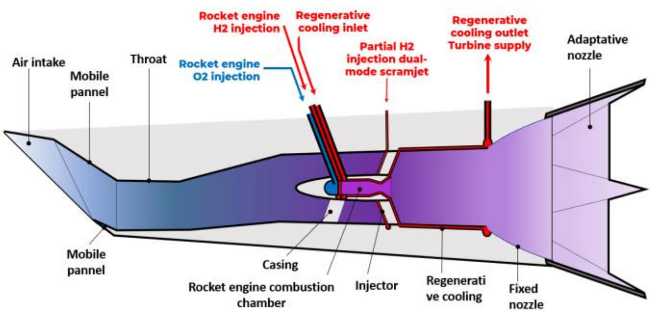

Figure 14: *Global engine configuration*

The initial iteration of the aircraft employed turbojet propulsion to achieve Mach 3. However, this approach was set aside, at least for now, due to the inadequate thrust generated for takeoff unless at least 2 units were used, which would result in excessive weight for a Single Stage To Orbit (SSTO) vehicle. During the early stages of development, a geometric arrangement idea was conceived to integrate the four propulsion modes (turbojet, ramjet, scramjet, and rocket engine) into a single system. However, designing a proprietary turbojet engine would require a substantial investment of time and money compared to utilizing an existing engine. Therefore, alternatives to the turbojet are being explored.

To determine the propellants to be used for the aircraft, a study was conducted with the aim of proposing at least four different propellant combinations. The evaluation was based on criteria such as ecological footprint, toxicity, specific impulse, mixture density, cost, and storage method. Combinations yielding a specific impulse lower than 350s were deemed insufficient for achieving SSTO capability and were not considered viable. The objective is to eventually study a version of the aircraft for each propellant combination to determine the most suitable option. Three possibilities currently available are the "traditional" propellant combinations: LOX/LH2, LOX/RP-1, and LOX/CH4. The last on selected is more exotic: LOX/C3H8 (propane). The aircraft version presented in this document utilizes the LOX/LH2 propellant combination.

Given the significant role of the air intake in ensuring proper engine performance, the selection of the intake type requires careful consideration. Therefore, a study was conducted to assess the effectiveness of a mixed compression intake. The efficiency of the intake with a given geometry was calculated using the article published by Bravo-Mosquera et al. (2016) [4], which utilizes oblique shock relations. The obtained efficiency is represented in red on Figure 15 and falls within the expected range. However, further detailed analysis will be necessary if this type of intake is selected. Additionally, a similar study is planned for a Busemann-type intake, commonly used for hypersonic missiles/aircraft, to determine the most suitable inlet type for the aircraft.

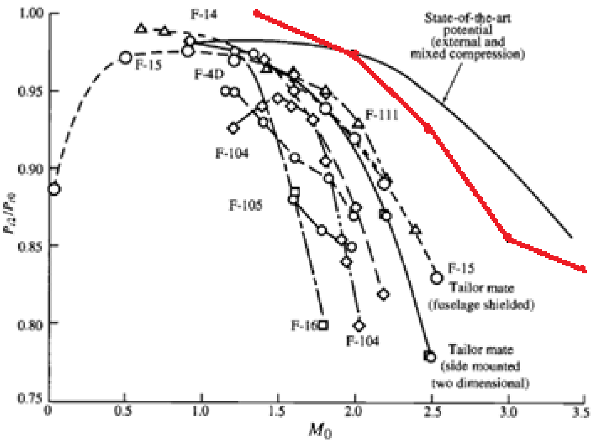

Figure 15: *air intake efficiency (red curve)*

Then, the established formulas were applied, incorporating the previously calculated intake efficiency, to determine the specific impulse of the dual-mode scramjet (Figure 16). It is observed that the calculated specific impulse falls within the typical range, which is reassuring regarding the accuracy of the calculation method.

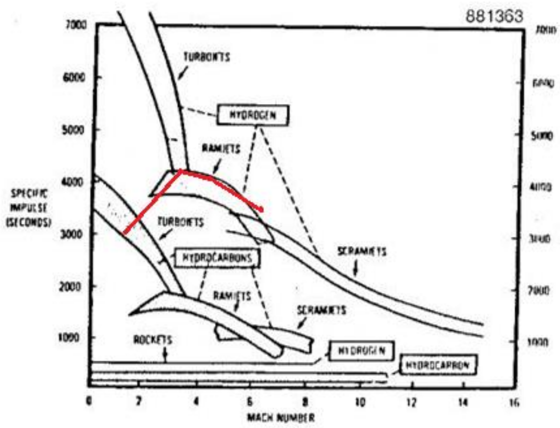

Figure 16*: dual-mode scramjet efficiency (red curve)*

It is important to note that the patented adaptive nozzle innovation is being utilized. This innovation enables the variation of the nozzle outlet, thereby increasing propulsive efficiency by adapting the nozzle outlet pressure to the ambient pressure. Also, the same engine (or rather the same nozzle) can then be used for the entire flight.

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Performance and Trajectory Consideration

To study the performance of each version of the aircraft produced, an attempt to reach orbit with each version is necessary. The final values for speed and altitude can then be recorded for comparison and analysis of each version's performance. The aim is to identify the causes of losses and find ways to improve future versions. Due to the unique characteristics of our atypical aircraft, existing code or software was not suitable, and a custom code needed to be developed.

The initial step in creating the code involved establishing the logic of its operation. Through a bibliographical research phase, we gathered the necessary elements, including some that were more advanced than the current stage of development. The chosen logic follows an explicit iterative process with adjustable time steps. Several functions were created, such as polynomials to characterize the atmospheric model within the 0 to 1000 km range, a function to calculate propulsive efficiency, and functions for reference point changes and aerodynamic coefficient calculations. The initial results obtained below highlight the challenges related to thrust and incidence.

At this stage, an initial set of thrust and incidence data was sought. The optimized trajectory of NASA's Hyperion project [1] was followed for this purpose. However, it is a challenging task to track a trajectory summarized by two variables: Mach number and altitude vs. time. The approach involved developing a customized proportional-integral-derivative (PID) controller to track the reference trajectory. The PID scheme is centered and utilizes altitude error and Mach number to determine the appropriate incidence and thrust, respectively. It is important to note that the PID parameters need to adapt as the flight conditions of the aircraft continuously change. With these modifications, the first trajectories have been obtained.

The following results correspond to VH2 with an initial mass of 42 tons (the latest studies show that we need to drastically increase propellant mass). Figure 17 presents a comparison between the trajectories of the Hyperion project and the aircraft. The PID controller demonstrates high effectiveness in tracking the reference trajectory. Divergence occurs when transitioning to the rocket engine phase. At this point, the objective shifts from following the Hyperion's curve to achieving precise performance. Specifically, the goal is to reach a 100 km orbit with a speed exceeding 7.9 km/s, which is the minimum orbital velocity. To analyze the behavior of the aircraft while following the determined trajectory, angle-of-attack and thrust profiles (depicted in Figure 18 and Figure 19, respectively) have been plotted. The observed variations correspond to changes in PID coefficients. To obtain more realistic flight conditions, additional smoothing techniques will be necessary. A redesign of the trajectory code is currently underway, to enable better trajectory studies with a spherical reference frame and code implementation. A better means of defining and following a trajectory once in orbit is to be added.

Finally, by comparing all the figures, the 50° angle-of-attack and 75 kN thrust (axial velocity of 8.1 km/s) are not yet sufficient to reach the final objective, as shown by the aircraft trajectory after 700 seconds.

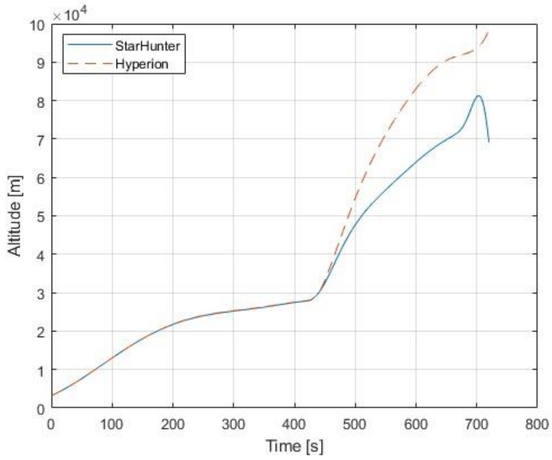

Figure 17: *Trajectories comparison of the Hyperion and the spacecraft*

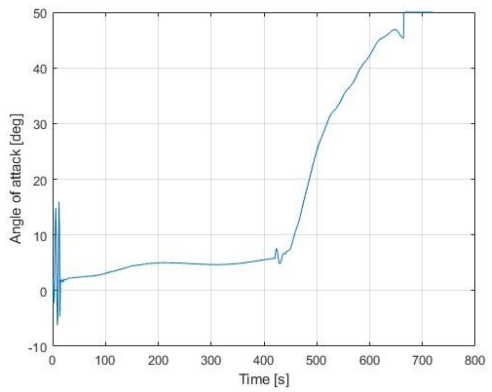

Figure 18: *Angle-of-attack profile*

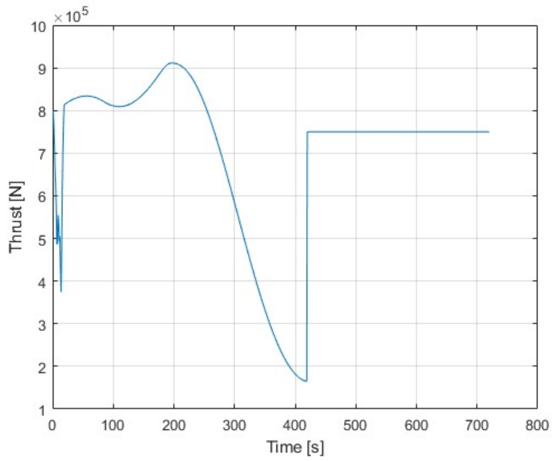

Figure 19: *Thrust profile*

Once trajectory performance has been achieved, and a high-performance calculation code is in place, a number of complementary studies are planned. These include aerothermal coupling. Of course, the aim of the trajectory is also to optimize weight and fuel consumption. To expedite the initial results concerning orbital performance, a spreadsheet

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has been implemented. It takes over the data from the conclusion of the air breathing propulsion phase and computes the final achievable mass at apogee along with the corresponding orbital velocity. The calculations are based on two thrust phases, each with its own angle. By considering the equations of motion and accounting for the gravitational work, it becomes possible to determine the apogee of the orbit that requires the corresponding orbital velocity to be reached. Although this method is less precise compared to the developed trajectory code, it still yielded results displayed in Figures 17 to 19 with an accuracy of less than 0.2 km/s. This discrepancy can be attributed to the absence of aerodynamic drag during the early stages of the rocket motor phase and the simplification of utilizing only two thrust angles. In order to estimate the air breathing phase, the necessary ΔV was extracted from the trajectory code, allowing for a more precise and proportionate calculation of the take-off weight. Overall, two types of fuels were investigated, namely liquid hydrogen and liquid propane.

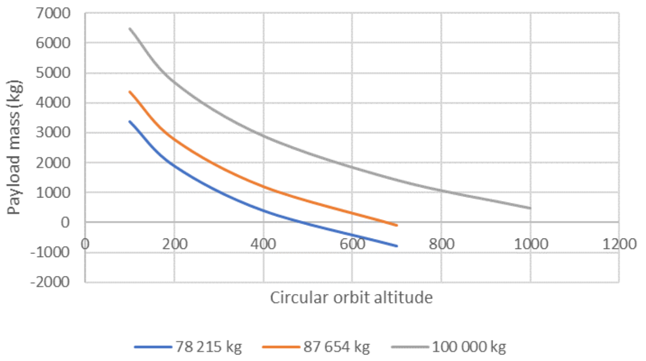

Figure 10 *: Final performances in function of take-off weight for LH2 fuel*

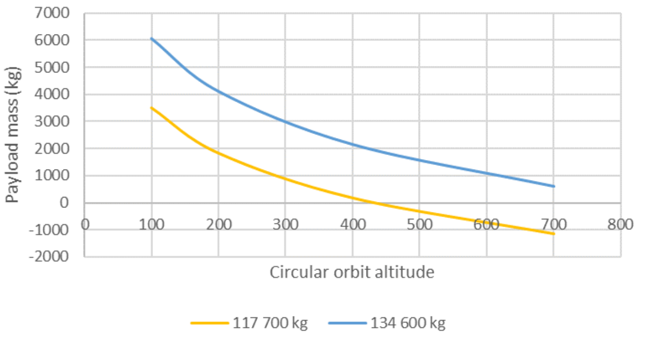

Figure 21*: Final performances in function of take-off weight for liquid propane fuel*

As can be seen in Figure 20, a hydrogen version would be capable of placing 3 tons at 400 km for a take-off weight of approximately 100 tons. By comparison, the performance of a mass-equivalent version with propane would only reach this level at 100 km apogee (Figure 21). On the other hand, it is clear that a vehicle tending towards 140 tonnes take-off weight would be able to match its hydrogen counterpart. In terms of volumes, the average propellant density of an LH2 version falls to 233 kg/m3, compared with around 1,000 kg/m3 for a propane version. Although it would be necessary to increase the vehicle's mass by 1.4 between LH2 and C3H8, it appears that the volume of this second version would be 3 times less. As propane is much less sensitive to storage conditions, it would be possible to convert each wing in a tank. It would therefore seem that a much better inert mass ratio is achievable with propane, whereas the ratio required for the LH2 juggles with the limits of current materials. Finally, as a performance enhancement, a propane version would certainly simplify development, ground and flight testing.

Safety studies

In order to maximize the safety and reliability of flights for our aircraft, numerous analyses are conducted to identify its strengths and weaknesses. The objective is to determine the strategies to be implemented to ensure that the "Manned Flight" criterion is integrated from the very beginning of the development process.

One of the essential tools is the Failure Modes, Effects, and Criticality Analysis (FMECA) in Table 4, which allows for the identification of potential failures and the evaluation of their impact. By classifying possible events based on their severity and probability, it becomes possible to assess the level of concern and determine the extent to which they need to be taken into consideration.

Numerical simulations and physical testing, to be conducted in the future, are also utilized to validate the performance of space systems. The implementation of this analysis has facilitated the breakdown of the system into major subsystems, which are further decomposed to analyze the criticality and probability of occurrence of each of these subsystems. This process has facilitated the identification of corrective actions to reduce the occurrence of these risks and establish a corresponding action plan.

Severity / Probability	Catastrophic (I)	Critical (II)	Marginal (III)	Negligible (IV)
Frequent (A)	High	High	Serious	Medium
Probable (B)	High	High	Serious	Medium
Occasional (C)	High	Serious	Medium	Low
Remote (D)	Serious	Medium	Medium	Low
Improbable (E)	Medium	Medium	Medium	Low

Table 4: *Example of Failure Modes, Effects, and Criticality Analysis (FMECA)*

The Hazard and Operability Study (HAZOP) has also been employed. This method is systematically used when there are high risks to identify, evaluate, and mitigate them if possible. The overall objective is to determine the weaknesses of current versions and how to overcome them (post-launch problems, failure to reach orbit, escape capsule cabin, etc.). Proposals to be discussed in the future.

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ECONOMIC AND COST ANALYSIS

Target markets

The two main versions of the spacecraft, namely the Crew and Cargo versions, have been designed to accommodate a diverse range of missions. These mission capabilities are expected to evolve in line with the increasing demands of the space industry. The primary performance objective is to transport either 4 passengers or 3 tonnes of payload to low Earth orbit, specifically up to 400 km, regardless of the orbit inclination. The main identified missions are summarized in Table 5 below.

Importance	Goal	Mission	Description
Main	Space exploration	Long-terms missions	*Several weeks (Crew) Several months (Cargo)*
		Astronaut and cargo transport	*To LEO (400 km) and space stations*
		Satellite launch, maintenance, and de-orbiting	And debris de-orbiting
Secondary	Return on investment	Space Tourism	Suborbital or orbital
		Military use	Observations or experiences
Hypothetical		Lunar lander	Use as a lunar lander

Table 5: *main tasks performed by the spaecraft*

Transporting astronauts and cargo

The company's vision is to expand space exploration missions and meet the increasing demand for sending astronauts, cargo, and other payloads into space.

Satellite launch, maintenance, and de-orbiting

In addition to human transportation, the spacecraft has the capability to deploy satellites. The concept involves offering satellite owners a comprehensive package that includes launch services, on-site technical assistance in the event of malfunctions, and proper de-orbiting at the end of the satellite's lifespan, in compliance with space regulations (LOS)

Long-duration missions

With its ability to accommodate human passengers, the spacecraft can support long-duration missions. This becomes particularly relevant with the decommissioning of the International Space Station (ISS) and the increasing demand for such missions. Short-duration missions could host up to 4 astronauts, enabling the execution of numerous experiments and allowing astronauts to return home in a less physically diminished state.

Space tourism

While not the primary focus of the company, the space tourism sector has been flourishing in recent years. As the spacecraft is capable of carrying astronauts, there is potential for commercial flights targeting the general public. Initially, suborbital flights could be offered, followed by orbital flights with the opportunity to spend 24 to 48 hours in space.

Military applications

There is a well-established interest from armed forces worldwide in shuttles capable of conducting long-duration missions in orbit. Countries such as the United States, China, and soon India employ discreet shuttle-like vehicles hidden under the nose cone of launchers for classified observation missions and zero-gravity experiments. In Europe, or more specifically in France, the spacecraft or its scaled-down versions could serve a similar purpose, eliminating the need for a separate launcher. The aircraft could remain in orbit for several months, offering military capabilities.

Lunar lander

Although purely speculative, the spacecraft has the potential to reach lunar orbit. This would require the implementation of in-orbit refueling, a technique that is currently under extensive research and development. By developing a vertical propulsion system, it may even be possible to consider lunar landing capabilities. However, it is important to note that these concepts are still in the realm of exploration and require significant technological advancements and further investigation.

Markets value estimations

According to [5], Europe could capture **€9.9 billion** by developing a manned space transportation system for LEO **between 2028 and 2040**. Global demand for manned transport to LEO is projected to increase from 30-50 seats in 2022 to 59-98 seats by 2040.

In 2022, the satellite market generated approximately **$10 billion** in revenue, with 90% of that coming from satellites weighing less than 2 tons, and 95% of those satellites being sent to LEO. Additionally, Europe has shown significant interest in future satellite maintenance technologies, particularly debris deorbiting. Failure to address the issue of space debris could result in a **loss of €1.5 billion** for the industry **by 2036**. [6]

A total of **€6 billion** will be dedicated to strengthening France's military space activities during the period of **2024-2030.**

The global lunar transportation market is estimated to be worth **$102 billion between 2020 and 2040** [7].

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Cost estimations

The cost analysis of the project was conducted using Koelle's TRANSCOST-MODEL (1984) [8], which encompasses both staged expandable vehicles and manned winged vehicles. This model utilizes a comprehensive database of space projects that were planned or executed during that time period to construct three sub-models. Figure 22 illustrates the structure of these sub-models, which are employed to estimate costs associated with development, production, and flight operations. However, considering that the proposed models are relatively outdated, a recalibration has been undertaken to account for current costs in the New Space era. This recalibration is based on the significant cost reductions observed in the past fifteen years, as reported by space launch companies.

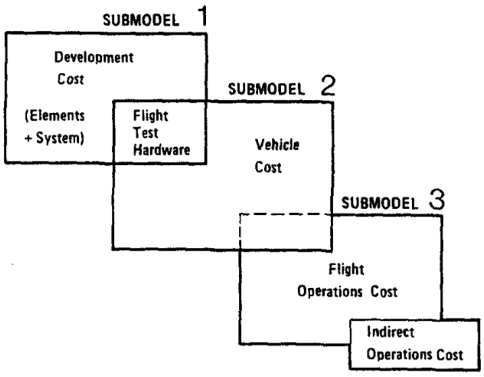

***Figure 22**: Cost of the spacecraft project*

In order to utilize the estimation method, assumptions need to be made regarding the vehicle, realization strategy, and final business model. Based on these assumptions, costs of vehicle development and propulsion can be estimated, considering the maturity of the technology and the company involved. The costs of the vehicle encompass both propulsion and the vehicle itself, taking into account the production rate. Flight operations costs (Figure 23) comprise direct costs (such as launch management, pre-launch operations, operations control, and propellants), the cost of replacing specific parts for reusable vehicles (refurbishment), and indirect costs (including administration, safety, engineering support, and maintenance of ground facilities). Additionally, there are charges to cover the development and production costs of the fleet in addition to these three types of costs. When studying project costs, two possible strategies could be considered.

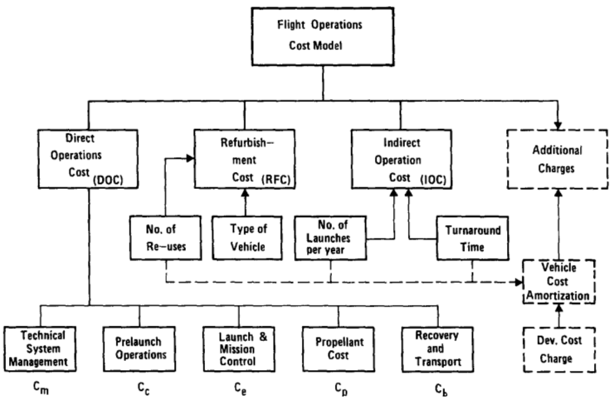

***Figure 23**: Organization of Flight Operations Costs*

(i) Strategy I

The strategy entails the production of a vehicle that has the capacity to transport 3 tons of payload, with an empty weight of 11 tons and a total weight of 100 tons. The assumptions made for the business model aim to ensure feasibility, with a projected development period of 8 years leading to the commencement of commercial flights in 6 years. The goals include establishing a fleet of 3 aircraft capable of conducting 15 flights per year, with a maximum of 45 flights per vehicle. The outcomes are presented in the Table 6 below.

Table 6 : Strategy I

Development Costs	**535,5 M€**
Vehicle	390,5 M€
Engine	145,0 M€
Production Costs	**30,86 M€**
Vehicle	27,8 M€
Engines (2 per vehicle)	3,06 M€
Flight Operations Costs (per flight)	**7,98 M€**
Direct Costs	**2,18 M€**
Technical system managment	(0,2 M€)
Prelaunch operations	(0,865M€)
Launch & mission control	(0,435M€)
Propellant	(0,680M€)
Refurbishment Costs	**0,562 M€**
Indirect Costs	**2,17 M€**
Added charges	**3,067 M€**
Development amortization	(2,38 M€)
Production amortization	(0,686M€)
Launch price (LEO – 400 km)	**2 660 €/kg**
Investment needed	**628 M€**

The estimations derived from Strategy I yield substantial development costs amounting to 535.5 million euros, with a unit cost of 30.9 million euros. These figures are factored into an economic model that incorporates a depreciation expense of 3.07 million euros per flight, in addition to flight operating costs. Consequently, the resulting consumer price stands at approximately 2,660 €/kg, which is considerably lower than the current market rates. Furthermore, the economic model does not consider the distinction between manned launches (higher profits) and payload launches. The calculated Internal Rate of Return (IRR) for the entire program is 7.8%. Considering the inherent risks associated with such a project, it is highly unlikely that this IRR would suffice to attract private investors like venture capitalists, who typically seek an IRR of 30% to justify undertaking such risks.

(ii) Strategy II

An alternative approach for achieving more cost-effective development is to adopt a second vision known as Design To Cost. This strategy revolves around a scaled-down model, operating at a 1:3 scale, with the capability to deliver 200 kg to Low Earth Orbit (LEO). The estimated take-off weight of this model is 10 tons, with an empty weight of 1.4 tons. The envisioned business model entails a development period of 5 years before initiating the first commercial flights. In this scenario, a fleet of 2 vehicles is utilized, with each vehicle

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conducting 25 flights per year, and the costs being amortized over a period of just 3 years.

Table 7 : Strategy II (part 1)

Development Costs	**200,6 M€**
Vehicle	162,6 M€
Engine	38,0 M€
Production Costs	**4,63 M€**
Vehicle	3,57 M€
Engines (2 per vehicle)	1,06 M€
Flight Operations Costs (per flight)	**4,03 M€**
Direct Costs	**1,46 M€**
Technical system managment	(0,183M€)
Prelaunch operations	(0,732M€)
Launch & mission control	(0,342M€)
Propellant	(0,097M€)
Refurbishment Costs	**0,454 M€**
Indirect Costs	**1,4 M€**
Added charges	**0,815 M€**
Development amortization	(0,661M€)
Production amortization	(0,154M€)
Launch price (LEO – 400 km)	**20 200 €/kg**
Consumer price	**25 000 €/kg**
Investment needed	**210 M€**

Strategy II, on the other hand, adopts a two-stage approach aimed at achieving the same outcome as Strategy I. The initial stage involves the development of a Minimal Viable Product (MVP), generating profits that will subsequently be reinvested in the development of a 1:1 version. The results from the first phase indicate a development cost of 200.6 million euros, taking into account the novelty of the technology and the initial product being developed. Positioned in the mini launcher market, this vehicle offers reusability and ease of use, resulting in a cost of just 20,200 euros per kilogram to reach Low Earth Orbit (LEO). Current market prices in the US range from 70,000 to 80,000 euros per kilogram, but many new players are targeting a range of 25,000 to 30,000 euros per kilogram. Based on this information, the proposed consumer price would be 25,000 euros per kilogram, aimed at generating initial profits for the company's further development. The calculated Internal Rate of Return (IRR) over the same overall period as Strategy I amounts to 16.4%, indicating positive progress, even if the decision is made to halt further progress at this stage.

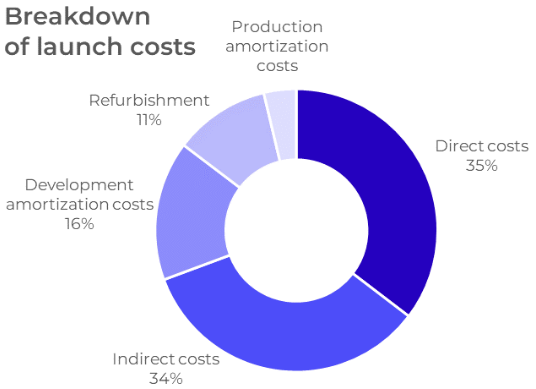

***Figure 24**: Breakdown of launch costs repartition*

According to Figure 24, a significant portion of launch costs, approximately 70%, can be attributed to direct and indirect expenses. This indicates that there is still potential for cost reduction. One area where cost savings can be achieved is through the utilization of a tarmac instead of a launch tower, which would result in a reduction in ground infrastructure expenses.

In the continuation of Strategy II, the focus lies on the development of the 1:1 scale vehicle, featuring a payload capacity of 3 tons for low Earth orbit, a take-off weight of 100 tons, and an empty weight of 11 tons. Development activities will commence once commercial flights of the 1:3 scale vehicle are initiated, benefitting from the generated revenues of 48.4 million euros. Since the 1:1 scale is essentially a scaled-up version of the 1:3 scale, and the underlying technologies are proven, the development timeline has been shortened to 3 years. This timeline allows for the establishment of a fleet consisting of 3 vehicles, each capable of conducting 15 flights per year. The cost estimate results for the second stage of Strategy II are presented in Table 8 below.

Table 8 : Strategy II (part 2)

Development Costs	**183,8 M€**
Vehicle	148,2 M€
Engine	35,6 M€
Production Costs	**27,94 M€**
Vehicle	24,88 M€
Engines (2 per vehicle)	3,06 M€
Flight Operations Costs (per flight)	**5,766 M€**
Direct Costs	**2,21 M€**
Technical system managment	(0,21 M€)
Prelaunch operations	(0,865M€)
Launch & mission control	(0,435M€)
Propellant	(0,700M€)
Refurbishment Costs	**0,625 M€**
Indirect Costs	**1,491 M€**
Added charges	**1,438 M€**
Development amortization	(0,817M€)
Production amortization	(0,621M€)
Launch price (LEO – 400 km)	**1 922 €/kg**
Investment needed	**122,5 M€**

The cost estimate results for Part 2 of Strategy II demonstrate a notable reduction in development costs, which were partially offset by the benefits gained from Part 1. Consequently, the cost per kilogram to Low Earth Orbit (LEO) has significantly decreased to 1,922 €/kg. These costs are considerably lower than the current market rates (8-10K€/kg), primarily due to the complete stage reusability and operational convenience. Assuming a consumer price of 4K€ /kg, the company's annual sales would reach 540 million euros. It is worth noting that these calculations do not take into account manned flights, which would further increase this figure. Overall, within the same project duration as Strategy I, Strategy II achieves an Internal Rate of Return (IRR) of 29%, aligning more closely with the expectations of Venture Capital in terms of the associated level of risk.

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Development plan

The development plan for the spacecraft project aligns with the current enthusiasm in the space sector. It acknowledges the need to carefully consider the different stages that will lead to the final aircraft. The task ahead is substantial, even for a craft of the spacecraft's size. The objective is to create an opportunity for the development of such an aircraft and take the initial steps as soon as possible, aiming to be at the forefront of the industry.

The proposed approach is to start the project independently and potentially collaborate with other players, while maintaining control, to leverage the expertise in aeronautics and space, especially from French and European sources

Long-term vision

The long-term development plan adopts an incremental approach, focusing on testing and validating the technological building blocks of the final aircraft. The plan involves the production of three types of prototypes, potentially on different scales. The specific number of prototypes at each scale is yet to be determined. The approach aims for cost-effectiveness, emphasizing the importance of low-cost prototypes and spare parts.

The planned key stages for the project involve the following sequential steps: developing a 1:8 scale non-propelled aircraft launched by a helicopter, creating a 35 kN rocket engine and conducting subsystem testing along with configuring it completely, constructing a 1:3 scale aircraft using the 35 kN engine to replace the APU at the rear, developing an airbreathing engine by utilizing the previous stage as a flying test bench and attaching interchangeable engines under the wings, achieving successful engine development and making attempts to reach orbit with the preceding version, advancing to the development of a 1:1 version based on the outcomes of the previous stage, and finally, endeavoring an orbital flight using the 1:1 version.

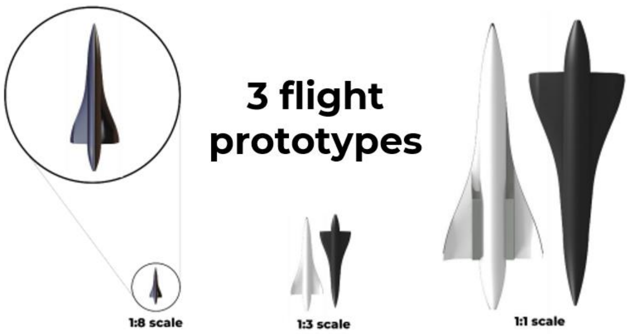

Figure 25*: Presentation of the 3 flight prototypes*

The Minimal Value Product (MVP) is a functional version of the 1:3 scale prototype capable of carrying out orbital missions, thereby generating revenue. This financial sustainability without reaching the final stage of the project is a valuable asset when it comes to attracting future investors. The ideal schedule estimates commercial missions for the 1:3 scale version by the end of 2027. The nominal version targets the first commercial flights at the beginning of 2029, while the degraded version is scheduled for the end of 2033. Modifications to the schedule are expected based on the results of the flight tests of the 1:3 scale version and the rocket engine.

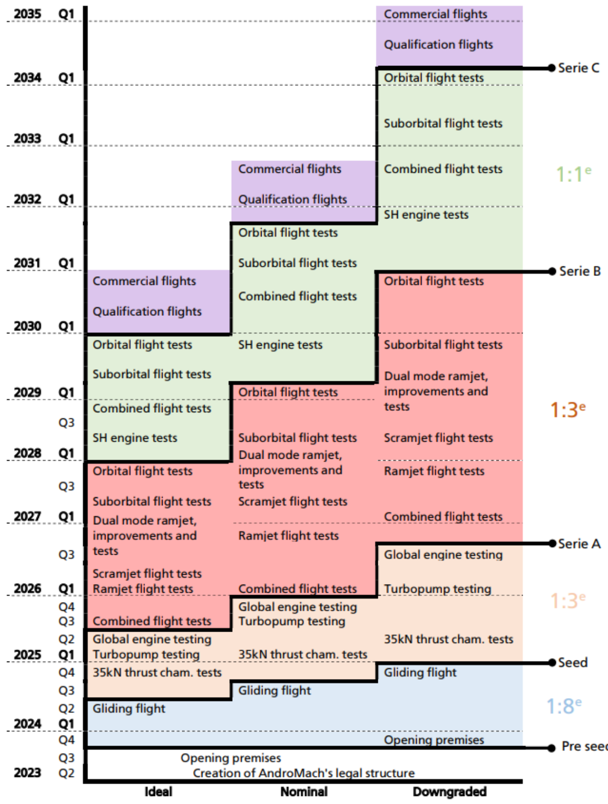

Figure 26*: Development plan*

Short-term vision

The short-term vision focuses on two developments: the glide system and the 35 kN rocket engine. According to the nominal schedule, these developments are projected to be completed by Q3 2024 and Q4 2025, respectively, with a little over one year between them.

- 1:8 scale aircraft

The initial step involves designing a non-powered aircraft on a 1:8 scale, which will be dropped from an altitude of 5000 m by helicopter. It will follow a predefined trajectory, perform a Side Roll manoeuvre, and land on the correct side of the runway. The flight will be managed by the on-board computer. This test will validate various technological building blocks, including the on-board computer, subsonic aerodynamic performance, external and internal structure design, Side Roll manoeuvre feasibility, landing gear integration, and the development of the ground electronics loop.

- 35 kN rocket engine

The 35 kN rocket engine plays a crucial role as it provides the first propulsion vector for the aircraft (1:3 scale). Placing the engine in the APU's position allows propulsion without significant modifications to the structure and avoids interfering with thefuture placement of the airbreathing engine under the wings. Successful validation of this stage would establish a foothold in the market and provide propulsion for flight testing in the next stage.

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SUMMARY

A study encompassing technical, organizational, and financial aspects of SSTO development has been presented to you. It provides an overview of the results obtained so far and the tools developed to achieve them. These tools allow for quick results and provide insights into the feasibility of proposed concepts.

The current studdy has shown that the latest concept studied, VH2, falls short of fulfilling its designated mission. However, similar to previous iterations of the aircraft (VH0 and VH1), the results obtained from VH2 have allowed us to refine our estimations, particularly in terms of onboard propellant mass and propellant choice between hydrogen and propane. This mass was previously underestimated, preventing the desired objective from being achieved. Therefore, the next iteration of the aircraft will take into account these more realistic onboard propellant masses, and the size of the aircraft will be modified accordingly to accommodate the updated volumes at play. The internal layout will also be modified in order to better balance the aircraft. The trajectory tool also needs to be completed to support the orbital insertion of the aircraft. Additionally, the calculations of thermal flux during ascent and atmospheric re-entry need to be finalized to determine areas requiring active cooling and to define the design of the thermal protection system.

In conclusion, although the goal of reaching LEO with a 3-ton payload has not yet been achieved, there is no indication that such a project is not feasible both technically and financially. Future iterations of the aircraft will be increasingly refined until reaching the version that appears to be the most optimal, which AndroMach will strive to develop.

ACKNOWLEDGEMENTS

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