Preliminary Design of a New Hybrid and Technology Innovative Suborbital Vehicle for Space Tourism

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The general enthusiasm aroused by space tourism combined with the great technological achievement of Scaled Composites with the SpaceShipOne in 2004 initiated a new era: suborbital space tourism. As of today, most of the vehicles have been designed for performance, combining the most advanced technologies from both aeronautics and astronautics. Nevertheless, in order to become viable, vehicles must be safe enough to carry paying passengers and they must match the increasing demand. Thus, the implementation of a new design process based on adapted requirements led to a new vehicle. The latter is mainly powered by newly designed hybrid rocket engines but it also makes use of turbofans for the first segment of the climb and a safe powered landing. It takes-off and lands horizontally and is able to carry up to eight passengers and two pilots to an altitude of 109 km. The micro-gravity experienced by the passengers lasts approximately 4 minutes while the maximum load factor is reduced to 3.3 g in order to improve the passenger experience.

I. Introduction

The first suborbital space commercial vehicle crossed the space border, defined by the Kármán line at 100 km (62 miles), in 2004. Even if the number of innovative public programs, started by Yuri Gagarin in 1961, tends to decrease, space tourism seems to be a promising future for manned spaceflights. Created in May 1996, the Ansari X Prize was a true catalyst to the development of reusable suborbital vehicles. This competition offered US $10 million to the first non-governmental organization that was able to “build and launch a spacecraft capable of carrying three people to 100 kilometers above the Earth’s surface, twice within two weeks”. The latter was won on October 4, 2004 by the company Scaled Composites thanks to its SpaceShipOne. This significant technological achievement is also supported by a growing demand for suborbital tourism. Many studies have been conducted to assess the medium and short-term suborbital market for space tourism. Results from surveys conducted by Airbus Group (previously EADS), Virgin Galactic and the Futron Corporation are illustrated on Figure 1. Since this new high-potential activity becomes real, a paradigm shift must occur. Indeed, rather than the only performance-related criteria such as maximum altitude and time in micro-gravity, the design process should focus more on safety requirements and passenger capacity. A typical suborbital mission profile includes a take-off using existing runways, a two-step climb, a micro-gravity parabola allowing the vehicle to reach the Kármán line, a reentry and a safe landing. The following characteristics have been chosen to increase safety: two pilots, horizontal take-off, hybrid rocket engine, automatic re-entry with low thermal and structural stresses and restart of turbofans for approach and landing. Finally, the increasing demand led to an 8-passenger vehicle. One of the requirements is to maximize the time spent in weightlessness as well as in space. The required minimum time in weightlessness is 200 s to match the performance of existing concepts. As the passengers comfort is one of the priorities, the maximum load factor allowed during the mission is fixed at 3.5 g. Since the maximum load factor for light aircraft in the regulation is 3.8 g, a 3.5 g load factor seems to be bearable without specific training.
The consideration of these new requirements demands the implementation of a complete design process. First, an energy based methodology for aircraft conceptual design adapted to suborbital vehicles will be presented. Then, since no historical data are available, a modeling and simulation environment will be created in order to size and integrate each subsystem. Finally, the performance of the designed vehicle will be computed in order to validate the aforementioned requirements.

II. Conceptual Design

The purpose of the conceptual phase is to determine a first estimate of the three main design variables: take-off gross weight $W_{TO}$, reference area $A$, and thrust at sea level $T_{SL}$. This section is mainly based on empirical models which require a general description of the vehicle to be used and so, some qualitative choices must be made. Since the vehicle operates at both low-speed and very high-speed, a double-swept delta wing seems to be the best alternative. Moreover, the latter provides a structural advantage for the reentry phase and the horizontal stabilizer becomes optional. A typical fuselage is designed based on the cockpit, cabin, tanks and other equipment constraints. Finally, since the concept must be able to take-off horizontally, two types of propulsion are used: a jet engine and a rocket engine. A low-bypass-ratio turbofan appears to be the best compromise since it has a reduced fuel consumption compared to a turbojet and a reduced weight penalty compared to a high-bypass-ratio turbofan. Afterburners and reverses can be added if needed for respectively take-off and landing. A hybrid rocket engine, which combines the two typical propulsion types by using a solid propellant with a liquid or gaseous oxidizer, is preferred. In addition to having an easily controllable thrust, hybrid engines are simpler, cheaper and more eco-friendly compared to liquid engines. They are also safer, more robust than solid engines, and may have higher specific impulses in theory.

A. Methodology

In order to find the key design variables, the sizing process described by D. Mavris\textsuperscript{9} and presented on Figure 2 is used. Historical data are used to build the three models: aerodynamics, propulsion and structure. Then, a loop between mission analysis and constraints analysis is initiated. The latter used weight ratios provided by the first one to output the two optimized scaling factors: thrust loading and wing loading. The latter are then used again by the mission analysis to determine the weight fractions. At convergence, the three key parameters are outputted. In this study, the hybrid engine is treated as a payload consumed during the rocket phase. Therefore, $T_{SL}$ only corresponds to the thrust required by the turbofans. To start the process, typical models presented by D. Raymer,\textsuperscript{10} R. Blanchard\textsuperscript{11} are used. Then, the mission and the constraints analyses are customized to match specific characteristics of suborbital missions.
B. Mission analysis

1. Take-off, climb and landing

The take-off phase is decomposed in three segments: ground acceleration, rotation and initial climb. Each segment is divided into small steps and Newton’s second law is applied in order to compute the fuel burned during each step. The detailed method and the expression of the different forces can be found in common aircraft design books.\textsuperscript{9,10,12,13} The idea is to compute at each step the ratio $\beta_i$ of the aircraft weight at the given step over its weight at the previous step (Equation 1). $T_i$ is the thrust and TSFC\textsubscript{i} the thrust specific fuel consumption during the step $i$ which lasts $\Delta t$.

$$\beta_i = 1 - \text{TSFC}_i \times T_i \times \Delta t$$

(1)

The same process is used for the jet-powered climb, the final approach and the landing phase. Thus, based on aircraft characteristics (aerodynamics and weight), the fuel burned throughout the mission is determined. Finally, this process ensures that the required fuel matches the available fuel.

2. Rocket climb and parabola

The climbing phase with the rocket engine is constraining for the vehicle structure because it endures a high load factor. The velocity, altitude and flight path angle when the rocket engine is shut down, at the burn-out point, will determine the length of the weightlessness phase and the maximum altitude reached during this phase.

The rocket climbing phase is divided into two parts: a rotation to reach a certain flight path angle followed by a zero angle of attack phase where the flight path angle will slightly diminish because of the weight. Then an unpowered phase begins when the rocket engine is shut down and the trajectory follows a parabola. Once these phases are modeled, an optimization program provides the best trajectory which minimizes the fuel consumption and maximizes both the duration of the weightlessness phase and the maximum altitude reached.

To model these phases, several hypotheses have been made for simplicity. A standard atmospheric model has been built to take into account the variations of the gravitational acceleration and of the air density with the altitude. The rocket phase starts at the end of the jet-propelled climb phase with a zero flight path angle, and the thrust of the rocket engine is considered constant during this phase. In fact, the thrust depends on the altitude but this dependence is only taken into account during the performance analysis. The chosen chamber pressure in the hybrid engine is 40 bar as
justified in Subsection III.C while the selected oxidizer to fuel ratio arises from the optimization performed by Rocket Propulsion Analysis (RP A), a design tool for the performance prediction of rocket engines. This software displays the curve of the specific impulse as a function of the altitude, used in the conceptual design part. Once the hybrid engine is shut down, the drag is neglected during the weightlessness phase. The Coriolis force is also neglected so that the trajectory remains in a plane. Since the difference with the true elliptic trajectory is only around 2.5%, it is assumed to be a parabola. The propellant consumption $\frac{dM_{\text{prop}}}{dt}$ is defined as a function of the specific impulse $I_{sp}$, the thrust $T$ and the gravitational acceleration $g_0$ by Equation 2.

$$\frac{dM_{\text{prop}}}{dt} = -\frac{T}{g_0 I_{sp}}$$

During the rotation phase, the vehicle flight path angle increases from 0° to a certain angle between 80° and 87°. To rotate, the vehicle can either increase its angle of attack, or benefit from a thrust vectoring nozzle. To get the most simple configuration for the rocket engine, the first option is adopted. Thus, the rotation is done by increasing the angle of attack, which increases the lift. The angle of attack is supposed to be controlled to remain constant during this phase. A numerical resolution of the equations of motion written in the aerodynamic frame is run by iterating on the time until the chosen flight path angle $\gamma_{\text{init}}$ is reached. The inputs of the program are the altitude and velocity of the vehicle at the beginning of the rocket climb phase, the flight path angle to be reached, the attack angle and other parameters which allow to calculate the air density and the reference area. The ratio $\beta = W(i)/W_{TO}$, which is the ratio of the weight at the iteration $i$ over the take-off weight is also calculated as well as the load factor $n$ with Equation 3. If the load factor is higher than the maximum load factor defined in the requirements, the case is eliminated.

$$n = \frac{\Sigma(\text{surface forces})}{W}$$

Once the chosen flight path angle is reached, the vehicle is brought to a zero angle of attack to cancel the lift. It continues to climb but the flight path angle will slightly decrease because of the weight. A numerical resolution is also used to solve the equations of motion with the same inputs as in the previous phase except the flight path angle. The iterations stop when the maximum load factor is reached. The rocket engine is then shut down to start the parabola. During the latter, the only force applied to the vehicle is its weight (drag is neglected) so that the trajectory becomes parabolic. In the conceptual design phase, the passengers are considered in weightlessness during all the parabola, until the vehicle comes back to the same altitude as at the beginning of the parabola. The atmospheric reentry is taken into account later during the performance calculation. A numerical resolution of the equations of motion is also performed since the gravity acceleration $g_0$ varies with altitude. The outputs are the weightlessness time (total time of the parabola), the time spent in space (at an altitude above 100 km), the maximum altitude reached, as well as the range of the parabola.

The trajectory during the rocket climb phase is optimized with respect to three criteria: propellant consumption (minimized), weightlessness time (maximized) and time spent in space (maximized). Many possible trajectories are obtained by varying the value of three parameters: the rocket engine thrust $T$, the flight path angle at the end of the rotation phase $\gamma_{\text{init}}$ and the angle of attack of the rotation phase $\alpha$.

$$\begin{cases} 1.5\ W_{TO} < T < 2\ W_{TO} \\ 80° < \gamma_{\text{init}} < 87° \\ 11° < \alpha < 15° \end{cases}$$

The thrust must be high enough compared to the vehicle weight but not too high because the propellant mass would then be too large. The flight path angle must be high enough, otherwise the thrust required to reach 100 km would be too large. The angle of attack must not exceed the critical angle of attack estimated at 18°. The different characteristics of the obtained trajectories are: thrust $T$, flight path angle at the end of the rotation $\gamma_{\text{init}}$, angle of attack during the rotation phase $\alpha$, propellant mass $M_{\text{prop}}$, duration of the weightlessness phase $t_{\text{ul}}$, maximum altitude $H_{\text{max}}$, range $L$, time in space $t_{\text{space}}$.

First, the trajectories which do not meet the aforementioned requirements are eliminated. Other criteria are also taken into account: the propellant mass must remain below 13,000 kg so that the take-off gross weight does not exceed 30 tons and the range of the parabola should not exceed 100 km to allow the vehicle to come back to its initial point. Among the remaining trajectories, the ratio $R$ is calculated for each trajectory (Equation 5) with $M_{\text{prop}}^{\min}$ the minimum propellant mass, $t_{\text{ul}}^{\max}$ the maximum weightlessness duration and $t_{\text{space}}^{\max}$ the maximum time in space.

$$R = \frac{M_{\text{prop}}^{\min}}{M_{\text{prop}}} + \frac{t_{\text{ul}}^{\max}}{t_{\text{ul}}} + \frac{t_{\text{space}}^{\max}}{t_{\text{space}}}$$
The coefficient $a$, $b$ and $c$ represent the importance of each optimization criterion. The propellant mass ($a = 7$) is chosen to be the most important compared to the other criteria ($b = c = 1$). The trajectory with the maximum ratio $R$ corresponds to the optimum trajectory.

**C. Constraints analysis**

In order to find the optimal point in terms of thrust loading and wing loading, an energy-based approach presented by D. Mavris\(^9\) is followed. Considering the aircraft as a point mass and assuming that the thrust is parallel to the drag (small angle of attack), Equation 6 can be derived. In this equation, $\alpha$ represents the thrust lapse which is a function of the Mach number and the engine setting, $\beta$ represents the instantaneous weight fraction which is the ratio of the weight at a given point in the mission over the take-off gross weight. Moreover, $C_D$, $K_1$ and $K_2$ are the drag polar coefficients so that $C_D = C_{D_0} + K_1 C_L^2 + K_2 C_L$. $\rho$ represents the dynamic pressure, $V$ the true airspeed and $R$ the additional drag due to the landing gear, ground friction, etc.

$$\frac{T_{SL}}{W_{TO}} = \frac{\beta}{\alpha} \left( \frac{q S}{\beta W_{TO}} \right) \left( \frac{n \beta W_{TO}}{q S} \right)^2 + K_2 \left( \frac{n \beta W_{TO}}{q S} \right) + C_D + \frac{R}{q S} + \frac{1}{V} \frac{d}{dt} \left( h + \frac{V^2}{2g_0} \right)$$  \hspace{1cm} (6)

Equation 6 is then applied for each constraint using the corresponding assumptions. Approach is treated differently since there is no thrust required for this phase. Instead, Equation 7 is used.

$$V_{ap} = k_{ap} \sqrt{\frac{2 \beta \rho C_{L_{max}}}{W_{TO}}} \frac{W_{TO}}{S}$$  \hspace{1cm} (7)

During take-off and landing, $R$ is described by Equation 8 and is otherwise set to zero. $C_{DR}$ is the drag coefficient augmented by the landing gear and the high-lift devices. Moreover, the climbing phase is assumed to be at constant speed with a load factor equals to one.

$$R = q C_{DR} S + \mu g \left( \beta W_{TO} - q C_L S \right)$$  \hspace{1cm} (8)

**D. Results**

This process has been implemented in Matlab with a loop between the constraint analysis and the mission analysis. At each iteration, the different coefficients are updated until convergence on the take-off gross weight. This process ensures that the fuel required corresponds to the fuel available and that the aircraft is able to meet all its requirements. Figure 4 shows the final constraint plot with the optimized design point. One can note that the constraints analysis does not concern the rocket phase since it has been completely treated in the mission analysis.
The design point is found by minimizing the thrust requirement while remaining in the feasible design space (displayed in white on Figure 4). The two constraining requirements are the take-off field length and the initial rate of climb. Since the mass is greatly reduced during the rocket phase, the approach phase is not constraining. This conceptual design phase provides $W_{TO} = 26$ tons, $S = 66.25$ m$^2$ and $T_{SL} = 12.1$ tons. These values must then be inputted in a modeling and simulation environment to provide a framework to the sizing process.

### III. Modeling and Simulation Environment

**A. Methodology**

In this section, every subsystem will be designed and sized while an iterative design loop will ensure the overall consistency. The sizing process, displayed on Figure 5 requires a modeling and simulation environment that takes as inputs the results of the conceptual design phase, the requirements and the mission description. The outputs of this section are converged values for dimensions, masses and layout of all subsystems as well as their integration into the vehicle. Since each of these subsystems impacts the others, a loop must be created until a converged configuration finally emerges. The following elements are considered in this section: wing, hybrid and jet engines, attitude control system, fuselage, empennage, control surfaces and thermal protections.

![Figure 5. Overview of the overall modeling and simulation environment](image-url)
B. Wing design

During the mission, the wing undergoes different types of phases with their own requirements: low speed take-off, supersonic flight, atmospheric reentry, etc. The take-off and the landing phases would suggest a small sweep angle and a small dihedral angle but a high aspect ratio as well as a relatively high thickness-to-chord ratio. However, the supersonic phase would prefer a small thickness-to-chord ratio and a high sweep angle in order to reduce the wave drag. Finally, the load that the wing undergoes during the reentry phase could be reduced by increasing the dihedral angle and reducing the aspect ratio. Therefore, the wing design is a trade-off between the various requirements imposed by the various phases of the mission. One of the best configuration seems to be a double-swept delta wing similar to the American Space Shuttle: a 81° inner sweep angle and a 45° outer sweep angle. Using Vorlat, a lattice vortex code provided by J. Marchman, Figure 6(a) provides a scaled view of the wing and Figure 6(b) provides the span-wise lift distribution. Once the wing has been scaled to match the conceptual design results, all dimensions can be found. The wing area is 66.6 m$^2$ and the span is 10.0 m. The root chord is 9.6 m and the tip chord 1.3 m. Finally, a NACA64-A airfoil has been selected since it is commonly used for supersonic aircraft.

![Figure 6. Results of the wing design](image)

C. Hybrid engine design

After a comparison between solid, liquid and hybrid systems, hybrid propulsion appears to be the best compromise between cost, simplicity, efficiency, safety and re-usability. In order to determine a preliminary sizing of the engine, the parameters found with the conceptual design process are used: propellant mass $M_{\text{prop}} = 11,892$ kg, thrust $T = 442,000$ N, combustion time $t_{\text{combustion}} = 76.9$ s, initial altitude $H_i = 11$ km, final altitude $H_f = 45.6$ km and the maximum engine diameter $D_{\text{max}} = 1.8$ m.

1. Sizing and mass budget

Because of performance requirements, the combustion chamber pressure is selected at 40 bar. The oxidizer is chosen to be nitrous oxide because of its multiple advantages (safety, cost, storage, self-pressurizing), without neglecting safety measures (to avoid decomposition hazards). The fuel is a paraffin based grain doped with aluminum. Research initiated at Stanford University shows that paraffin based fuels have higher regression rates compared to conventional fuels because of the presence of a thin liquid layer during combustion. In order to estimate the performance parameters of the engine, empirical coefficients for an ONERA (French Aerospace Laboratory) fuel are used. After sizing each component of the engine, empirical equations can be used to determine their approximate mass (nozzle, chamber, oxidizer tank, pressurization tank, propellants), with a methodology suggested by R. Humble.

**NOZZLE DESIGN:** An important factor to determine is at which point in the flight the nozzle will be fully expanded. Using RPA, different configurations are tested in order to optimize the specific impulse over the entire flight, roughly between 11 and 45 km (Figure 7). This analysis enables to determine the altitude of optimal expansion before flow separation occurs. In this case, the nozzle is fully expanded at 18.6 km and the exit pressure is $p_e = 0.0069$ bar. Using the outputs of RPA (Mach, thrust coefficient) and the design requirements, the throat diameter and the expansion ratio...
can be determined. A bell nozzle is a good compromise between length and efficiency. Because a full design of the nozzle is very tedious, empirical charts are used in order to get a preliminary design of the nozzle. Its length is chosen to be 80% of a conical nozzle with a 15° half-angle. Phenolics, a composite material commonly used for ablative nozzles, is selected and R. Humble\textsuperscript{16} suggests Equation 9 to calculate its mass.

\[ m_{\text{nozzle}} = 125 \left( \frac{M_{\text{prop}}}{5,400} \right)^{\frac{1}{3}} \left( \frac{A_e}{10.1} \right)^{\frac{1}{3}} \]  \hspace{1cm} (9)

**Combustion Chamber Design:** In order to avoid shocks in the chamber, a safety coefficient of 1.6 is used to determine the inner port radius. The oxidizer to fuel ratio (O/F) shifts as the fuel regresses but the chamber can be designed so that the average O/F is close to the optimum value (9.4). Using the propellant mass, the oxidizer and fuel masses can be computed. Choosing a high regression fuel enables to design an efficient single port design. In order to determine the outer diameter of the fuel grain, the regression equation for hybrid rockets is used (Equation 10).

\[ \dot{r} = aG_{oz} = a \left( \frac{D_{oz}}{\pi r^2} \right)^n \]  \hspace{1cm} (10)

Using the fuel density and the geometrical properties, the length of the chamber can be found. In the preliminary design phase, the pre and post combustion chambers are sized with the empirical Equation 11.

\[ \frac{L_{\text{pre-chamber}}}{D_{\text{fuel}}} = 0.6 \]  \hspace{1cm} (11)

In order to simplify the chamber, only the fuel grain, the thermal protection and the casing are taken into account. A Polyethylene thermal protection was suggested by ONERA for its low weight and cost. Because a thermal analysis must be done in order to determine the thermal protection thickness, a largely overestimated thickness is assumed at this stage (\(c_{PE} = 2\) cm). A composite including graphite fibers is selected for the casing in order to lighten the structure of the engine. Using hoop stress calculations in order to ensure that the casing holds the required pressure, the thickness and consequently its mass can be found. Note that a safety factor of 1.5 is used in these calculations.

**Oxidizer Tank Design:** A cylindrical tank with hemispherical end caps is the most common design for oxidizer tanks. It can be sized using the density, the oxidizer mass and the maximum diameter constraint. The thickness of the pressure vessel is calculated for a pressure of 60 bar. Considering different geometries, the volume varies considerably, as does the mass of the tank, in composite as well. 20% ullage is included in the design.

**Pressurization System Design:** The advantage of using nitrous oxide is the fact that it is a self-pressurizing gas. Nevertheless, blow down of the tank prevents the system from having a constant oxidizer mass flow rate without throttling. Moreover, the performance of the engine is drastically different when the oxidizer is in liquid or gaseous phase in the oxidizer tank. Pressurizing the system increases robustness because the transition can be more easily...
determined. Blow down of the tank requires further analysis and would reduce the overall efficiency. A helium pressurization system is most convenient in this case. Using common values for the final pressure of the pressurant vessel, the desired oxidizer tank pressure and the same diameter constraint, the initial pressure that is required can be determined and the system can be sized at a preliminary stage. In the case of a spherical tank the required pressure is very high (341 bar), implying a very thick wall and heavy materials. The volume to weight factor could be optimized in more depth. For simplicity, we assume the use of a spherical helium tank in titanium. Note that because of the size of this pressure vessel, this would be a high cost option.

**INJECTOR:** Although the injector, chosen in aluminum, has a small impact on the total mass budget of the engine, the empirical Equation 12 from R. Humble is used. \( r_f \) is the final port radius of the fuel grain and \( \rho_{alu} \) the density.

\[
m_{\text{inj}} = 0.025 \rho_{alu} \pi r_f^2
\]

2. Configuration optimization

The total propellant mass flow \( D_{tot} \) which is the ratio of the total propellant mass over the combustion time seems very high for this type of engine. In particular, the oxidizer mass flow required in order to meet the performance is particularly high. One solution to reduce these values is to have multiple combustion chambers. The main advantage is that the length would be considerably reduced. However, the structural mass would be significantly higher and this solution would add extra difficulties such as multiple valves and ignition systems. Another solution is to reduce the length of the two tanks by having a simple tank with two embedded reservoirs as used in the Centaur rocket stage. We will refer to this solution as the “compact vessel design”. However, because the properties of the tank walls are very different for these pressure vessels, the two tanks would be completely separate, and this solution would only reduce the lost space caused when assembling a spherical tank and a cylindrical tank with hemispherical end caps, referred as “conventional vessels”.

A total of four configurations are analyzed: compact vessels with 1 chamber, compact vessels with 3 chambers, conventional vessels with 1 chamber and conventional vessels with 3 chambers. The sizing of these four engine designs are determined and a mass budget is computed for each configuration. Again, a 10% margin is added to account for the multiple subsystems that were not analyzed at this point of the design process. The longer the engine, the lighter it is. The more compact cases have a higher total mass. Because both mass and length of the engine are to be reduced, a simple optimization process is used. The factor \( K \) is introduced by Equation 13 with \( M_s \) the structural mass of the engine, \( M_{min} \) the minimum structural mass out of the four configurations, \( L_{tot} \) the total length of the engine, \( L_{min} \) the minimum length out of the four configurations. Multiple values of \( \lambda \) are tested and the best compromise is the configuration using the conventional vessel design and 3 combustion chambers with separated nozzles. Table 1 sums up the principle dimensions of the engine.

\[
K = \lambda \frac{M_s}{M_{min}} + (1 - \lambda) \frac{L_{tot}}{L_{min}}
\]

<table>
<thead>
<tr>
<th>Spherical Helium tank</th>
<th>Diameter (m)</th>
<th>1.75</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \text{N}_2\text{O tank} )</td>
<td>Diameter (m)</td>
<td>1.75</td>
</tr>
<tr>
<td></td>
<td>Volume (m(^3))</td>
<td>10.46</td>
</tr>
<tr>
<td></td>
<td>Length cylindrical part (m)</td>
<td>3.18</td>
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<tr>
<td></td>
<td>Total length (m)</td>
<td>4.94</td>
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<tr>
<td><strong>Combustion chamber</strong></td>
<td>Initial port radius (m)</td>
<td>0.128</td>
</tr>
<tr>
<td></td>
<td>Final port radius (m)</td>
<td>0.4</td>
</tr>
<tr>
<td></td>
<td>Fuel grain length (m)</td>
<td>0.7</td>
</tr>
<tr>
<td></td>
<td>Total length (m)</td>
<td>1.22</td>
</tr>
<tr>
<td><strong>Nozzle</strong></td>
<td>( A_{\text{throat}} ) (m(^2))</td>
<td>0.020</td>
</tr>
<tr>
<td></td>
<td>( A_c/A_t )</td>
<td>26.8</td>
</tr>
<tr>
<td></td>
<td>Length (m)</td>
<td>1.25</td>
</tr>
<tr>
<td><strong>Total engine length (m)</strong></td>
<td></td>
<td>9.81</td>
</tr>
</tbody>
</table>

Table 1. Sizing - Conventional vessels, 3 chambers
3. **Thrust determination**

Once the design of the hybrid engine is complete, its thrust can be drawn as a function of altitude $h$ with Equation 14 (Figure 8).

$$T(h) = I_{sp}(h) \times D_{tot} \times g_0$$

(14)

The values of the specific impulse as a function of the altitude are given by the software RPA, and a mean value of the mass flow rate $D_{tot}$ is used.

![Figure 8. Thrust of the hybrid engine as a function of altitude](image)

4. **Hybrid design conclusion**

A hybrid engine was chosen for this mission. Although hybrid propulsion is not commonly used for launching applications, many projects today focus on this technology. A preliminary design of an engine that meets the mission requirements was determined. This design only focuses on four main components of the engine: the pressurization system (spherical titanium tank), the oxidizer tank (composite cylindrical tank with hemispherical end caps), the combustion chamber (3 cores) and the nozzle (3 bell shaped nozzles). Figure 9 presents the 3 view drawing of this engine. Using composites for both the oxidizer tank and the chambers, instead of typical materials such as aluminum considerably reduces the mass of the engine. This type of structure is currently used for suborbital missions by Scaled Composites and therefore demonstrates its feasibility and readiness. The final design is presented in Table 2 in terms of material, thickness, density and mass. The mass distribution between engine structure and fuel is also provided.

![Figure 9. 3 view drawing of the engine](image)
<table>
<thead>
<tr>
<th>Component</th>
<th>Material</th>
<th>Thickness (m)</th>
<th>Density (kg/m³)</th>
<th>Mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Helium tank</td>
<td>Titanium</td>
<td></td>
<td></td>
<td>912</td>
</tr>
<tr>
<td>( \text{N}_2 \text{O} ) tank</td>
<td>Composites</td>
<td>0.012</td>
<td>1,550</td>
<td>498</td>
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<tr>
<td>Injector</td>
<td>Al 2219</td>
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<td>2,800</td>
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<tr>
<td>Thermal protection</td>
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<td>Chamber casing</td>
<td>Composites</td>
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<td>1,550</td>
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<tr>
<td>Nozzle</td>
<td>Phenolic-based</td>
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<td>389</td>
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<tr>
<td>Propellants</td>
<td></td>
<td></td>
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<td>11,892</td>
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<tr>
<td>Helium</td>
<td></td>
<td></td>
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<td>182</td>
</tr>
</tbody>
</table>

\[ \text{Structural mass} = 2,539 \]

\[ \text{Total engine mass} = 14,434 \]

\[ \text{Structural mass coefficient} = 0.18 \]

Table 2. Mass budget using composite pressure vessels

D. Turbofan design

Based on the results from the conceptual design, the turbofans require a thrust at sea level equal to \( T_{SL} = 122 \text{ kN} \). Moreover, since the most constraining phases of the flight (take-off and initial climb) only last a short period of time, afterburners are chosen. The M88 from Snecma which already equips the Rafale from Dassault-Aviation can be used with a scaling factor of \( \lambda = 0.89 \). The latter is directly used to find the mass and the mass flow of the new engine and its square root is used for the dimensions. Table 3 presents the scaled parameters of each of the two turbofans based on data from J. Desclaux.\textsuperscript{17} Finally, these turbofans are placed on the fuselage since the bottom of the wing must be preserved due to intense constraints during the reentry phase.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust with afterburners</td>
<td>67.0 kN</td>
</tr>
<tr>
<td>Thrust without afterburners</td>
<td>44.7 kN</td>
</tr>
<tr>
<td>Length</td>
<td>3.34 m</td>
</tr>
<tr>
<td>Inlet diameter</td>
<td>0.66</td>
</tr>
<tr>
<td>TSFC with afterburners</td>
<td>0.80 kg/daN.h</td>
</tr>
<tr>
<td>TSFC without afterburners</td>
<td>1.7 kg/daN.h</td>
</tr>
<tr>
<td>Mass flow</td>
<td>0.71 kg/s</td>
</tr>
<tr>
<td>Turbine inlet temperature</td>
<td>1,850 K</td>
</tr>
<tr>
<td>Bypass ratio</td>
<td>0.3</td>
</tr>
<tr>
<td>Weight</td>
<td>801 kg</td>
</tr>
</tbody>
</table>

Table 3. Engine characteristics (per engine)

E. Design of the attitude control system

Once in space, the vehicle cannot be controlled by aerodynamic surfaces because the atmosphere is too thin. A three-axes control is necessary to allow independent motions around each axis. Indeed, roll control is needed for passengers to see the Earth during the parabola, and the pitch and yaw controls are required to prepare the reentry. Attitude control engines are chosen based on the vehicle weight, the desired rotation rate, and the capabilities of existing attitude control systems. Two engines provide a torque in each direction to allow control around each axis. To increase reliability, these six engines are mounted on gimbals.

I. Requirements identification

First, the torque \( T \) must be determined using Equation 15 where \( \omega \) is the rotation speed and \( I \) the matrix of inertia. The matrix \( I \) is assumed to be diagonal because of negligible non-diagonal terms. It is also assumed that \( I_y = I_z \), confirmed by the matrix obtained with the mass budget.

\[ T = I \omega + \omega \otimes (I \omega) \]  \hspace{1cm} (15)

The rolling moment is sized to allow the vehicle to perform a complete turn with a speed of \( 4^\circ/\text{s} \). No rolling motion is considered during the reentry because thermal protections would not withstand the heat in every part of the vehicle.
Calculations demonstrate that a torque of 100 Nm applied during 22 s allows a complete turn in approximately 2 min. Thus, up to three turns can be achieved during the weightlessness phase.

The pitching moment is sized to decrease the pitch attitude from 80° to -20°. By applying a torque of 400 Nm during 32.6 s, the 100° rotation is achieved with in approximately 1 min 40 s with a speed of 2°/s. Since lateral motion is not considered during reentry, it is assumed that the requirements are similar for the yawing moment.

In order not to weaken the thermal shield under the vehicle and to benefit from the highest level arm, the yaw and pitch engines are located in the nose and the tail while the roll engines are placed close to the wing tips. Finally, the thrust required by each engine is calculated based on the aforementioned torques and positions. Applying a 20% margin, it is found that the two roll engines must provide a thrust of 25 N during 56.1 s. The two pitch and the two yaw engines should deliver a thrust of 80 N during 80 s.

2. Design

Usually, liquid engines are used for both small and high thrusts whereas cold gas engines are only used for thrusts lower than 100 N.

Usually, liquid engines are used for small to high thrust, and cold gas engines for thrust levels lower than 100 N. In this case, cold gas thrusters are appropriate for this range of thrust, and it is a safe and simple system. The gas is stored in a high pressure tank. A regulator reduces the gas pressure before it is accelerated in a nozzle to generate thrust. To size the attitude control thrusters, the equations and principles are described by R. Humble.\textsuperscript{16} Helium is selected as the pressurized gas because of its low reactivity and relatively high specific impulse (around 180 s). The following describes the sizing of one of the pitching thrusters, for a thrust \( F = 80 \) N during 80 s. A safety margin of 50% is taken so that the thruster is designed for a combustion time \( t = 120 \) s.

A first design is performed assuming a negligible difference between the exit pressure \( p_e \) and the atmospheric pressure \( p_a \). By using the specific heat ratio and the gas constant of Helium, its characteristic velocity \( c^* \) can be calculated. With the mass flow rate \( \dot{m} \) given by Equation 16, the thrust and the ratio \( p_e/p_a \), the specific impulse is plotted as a function of the nozzle area ratio \( \epsilon \) on Figure 10. \( p_r \) is the regulation pressure.

\[
\dot{m} = \frac{A_0 p_r}{c^*} \quad (16)
\]

The specific impulse is nearly constant for a nozzle area ratio greater than 25 so that a value 28 is chosen for this thruster, which gives a specific impulse of 175 s, an exit Mach number of 7.3 and a pressure ratio of 64.10\textsuperscript{-5}. The mass flow rate is then 0.0466 kg/s for a thrust of 80 N. To determine the nozzle dimensions, the curves of the exit area and the throat area as a function of the regulation pressure are drawn. To have reasonable dimensions for the nozzle, the regulation pressure is fixed at 13 bar, for a thrust and an a nozzle-exit diameter \( D_t = 0.007 \) m and \( D_e = 0.037 \) m. A high-efficiency bell nozzle is selected with \( \lambda = 0.98 \) and an initial tank pressure \( p_{in} = 400 \) bar.

To predict the performance of the thruster, the real thrust is calculated without neglecting the exit pressure of 835.15 Pa. The thrust obtained is 79.29 N, calculated using the nozzle efficiency \( \lambda \). It is very close to the required value. The new value of the mass flow rate is 0.466 kg/s. The total used propellant mass is then \( m_p = 5.58 \) kg. To obtain the total propellant mass required, including the propellant mass left over \( m_r \) when the tank pressure gets down to the regulator pressure, the system of equation to be solved is (with \( T_0 \) the gas temperature and \( V \) the tank volume). The tank volume is then 0.089 m\textsuperscript{3} and the residual propellant mass is 0.187 kg.

\[
\begin{align*}
& p_r V = m_r RT_0 \\
& p_{in} V = (m_r + m_p) RT_0
\end{align*}
\quad (17)
\]

To calculate the mass budget, a bell nozzle is chosen (with a half-angle of 15° for a conical nozzle and a ratio of 0.675 for the bell nozzle). The length obtained is 0.0379 m. Phenolics are also used, similarly to the hybrid engine nozzle. Its mass is calculated with Equation 18 suggested by R. Humble.\textsuperscript{16}

\[
m_{noz} = 0.256 \times 10^{-4} \left[ \frac{(m_{prop} c^*)^{1.2} \epsilon^{0.3}}{p_r^{0.8} \tan(15)^{0.4}} \right]^{0.917} 
\quad (18)
\]
Using a spherical gas tank in titanium, the helium tank radius $r_t$ is then 0.277 m for this thruster and has a mass of 28.7 kg. A total mass of 15 kg is considered for all the other elements: pipes, regulator, valve.

The same process is applied to design the other thrusters (rolling and yawing control thrusters). Table 4 displays the results.

<table>
<thead>
<tr>
<th></th>
<th>$D_t$ (m)</th>
<th>$D_e$ (m)</th>
<th>$r_t$ (m)</th>
<th>$m_{gas}$ (kg)</th>
<th>$m_{noz}$ (kg)</th>
<th>$m_{tank}$ (kg)</th>
<th>$m_{other}$ (kg)</th>
<th>$m_{tot}$ (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pitching</td>
<td>0.007</td>
<td>0.0371</td>
<td>0.277</td>
<td>5.767</td>
<td>0.089</td>
<td>28.68</td>
<td>15</td>
<td>49.53</td>
</tr>
<tr>
<td>Rolling</td>
<td>0.0039</td>
<td>0.0208</td>
<td>0.167</td>
<td>1.267</td>
<td>0.0204</td>
<td>6.302</td>
<td>15</td>
<td>12.58</td>
</tr>
<tr>
<td>Yawing</td>
<td>0.007</td>
<td>0.0371</td>
<td>0.280</td>
<td>5.973</td>
<td>0.0909</td>
<td>29.7</td>
<td>15</td>
<td>50.76</td>
</tr>
</tbody>
</table>

Table 4. Dimensions and masses of the different attitude control thrusters

Considering there are three thrusters in the nose, a common gas tank is used to reduce the required space and weight. Thus, by adding a 10% margin, the working propellant mass is 26 kg, the structure mass is 315 kg and the thrusters mass is 350 kg. In addition to the propulsive system, a 3-axes inertial sensor is necessary to track the angular speed of the vehicle. As for the attitude, Earth and Sun sensors are used. Finally, a mass of one ton is allocated to the attitude control system.

**F. Fuselage design**

The fuselage must meet different requirements coming from multiple disciplines. Indeed, it must safely carry the passengers, the landing gear as well as the fuel tanks and miscellaneous equipment. It must also provide an ideal working environment for the pilot and a moment arm large enough for the empennage while providing a structural integration for the entire aircraft.

According to M. Sadraey’s historical data\(^{13}\) for fighter aircraft, a 50 cm nose should be enough. However, additional space is required for the cold gas engines and tanks. The length of the nose is therefore fixed at 80 cm which provides a volume approximately equal to 1 m\(^3\).

Using the requirements for fighter aircraft cockpits provided by M. Sadraey,\(^{13}\) the designed cockpit is displayed on Figure 11. Similarly, the cabin is sized using typical economy class seat dimensions but with more space between seats. Figure 12 shows a vertical view of the cabin.

![Figure 11. Cockpit](image)

![Figure 12. Vertical view of the cabin](image)

The nose landing gear room is under the cockpit and the main landing gear room is under the propulsive system. Moreover, according to historical data,\(^ {13}\) all the other subsystems can be grouped together in a single portion of the fuselage with a required volume corresponding to a 50 cm section of the fuselage. Finally, the length of the jet-fuel tanks are given by Equation 19 where $\rho_{ca}$ is the density of the fuel, $W_{ca}$ its required weight and $R_{ca}$ the radius of the fuel tanks.

$$L_{ca} = \frac{W_{ca}}{g_0 \pi R_{ca}^2 \rho_{ca}}$$

(19)
G. Empennage design

Since the vehicle has a delta wing, the only required stabilizer is the vertical one. The latter is placed at the rear of the fuselage. M. Sadraey provides a design methodology based on the volume coefficient $V_V$ defined by Equation 20. In this equation, $S_V$ is the vertical stabilizer area, $b$ the span and $l_V$ is the distance between the aerodynamic center of the vertical stabilizer and the aerodynamic center of the wing. A typical value for such vehicles is $V_V = 0.045$. Moreover, the aspect ratio is fixed at $AR_V = 1.3$ and the sweep angle at $\phi_V = 40^\circ$ by comparison with other fighter aircraft.

$$V_V = \frac{l_V S_V}{b S}$$  \hspace{1cm} (20)

H. Landing gear design

A standard tricycle configuration is chosen for the landing gear. The position of the landing gear is constrained by various requirements such as stability, ground clearance and maneuverability. The latter are fully described by M. Sadraey\(^{13}\) who also provides an empirical relation between $Y_r$ the distance between the two main landing gears and $H_g$ the height of the center of mass of the overall aircraft (Equation 21). Moreover, approximately 80\% of the weight must be carried by the main landing gear. This weight distribution combined with the aforementioned constraints are included in the overall design process loop.

$$\arctan\left(\frac{Y_r}{2H_g}\right) \geq 25^\circ$$  \hspace{1cm} (21)

I. Control surfaces design

The control surfaces are sized using XFLR5,\(^{18}\) an analysis tool for airfoils, wings and planes based on the Lifting Line Theory, on the Vortex Lattice Method, and on a 3D Panel Method. The hereby method enables flexibility in terms of angle of attack and sideslip angle for different configurations. The surfaces are first designed and analyzed before the associated wing is simulated. The process is iterative until the expected criteria are met.

1. Modeling

First of all, the vehicle must be modeled in a Computer Aided Design (CAD) software with its main parts: NACA64-A wing, fuselage and stabilizer (Figure 13). The different masses given by the mass breakdown are positioned on the model to enable computation of the inertia and the center of mass of the vehicle. The moving part is defined in terms of chord and span.

![Figure 13. Complete model in XFLR5](image)

2. Performance analysis

Once the geometry is defined, the aerodynamic characteristics must be obtained. The drag polar of the new airfoil is hence computed for each position of the moving part.
If the control surface to study is an aileron, the wing must be modified and the same method is applied for the stabilizer. For each section of the wing, the airfoil (with moving part or not) is defined and enables the creation of a control surface for the vehicle. At this step the span of the control surface is defined.

The characteristics of each configuration can now be computed varying the angle of attack or the sideslip angle. The different stability and control aerodynamic coefficients are obtained as a function of the angle of attack and the deflection of the control surface.

3. Results

The stability and control aerodynamic coefficients obtained are used in a simulation software such as Flight Gear to test the behavior of the designed configuration. The previous steps are iterated until correct dynamics of the vehicle are obtained (response time, stability, etc). Finally, the aileron dimensions represent 21% of the wing chord and 41% of its span while the rudder dimensions represent 35% of the vertical stabilizer and 67% of its span.

J. Structure design

One of the main drivers in the choice of the materials is to minimize the mass while keeping properties of stiffness and resistance. Thanks to the graphs given by M. Ashby, different materials have been studied. Magnesium, even if its density is lower, was eliminated because of its low Young modulus. Then, despite its strong resistance and high Young modulus, titanium was also eliminated since its density is too high, and the vehicle would be too heavy. However, it is used for secondary structures. Thus, CFRP (Carbon Fiber Reinforced Plastic) was chosen because of both its improved weight/resistance ratio compared to aluminum and its good thermal resistance. In addition, increasing research is performed on composite materials so that they will be improved even more.

1. Choice of the type of structure

Compared to composite sandwiches which offer a great stiffness, a more rigid composite thin skin with stiffeners is selected. Several arguments stood out such as flexibility in maintenance and design of the structure, even if the skin provides less protection against debris.

2. Wing study

Wing is whether structured with tilted spars (compared to the wing direction) or straight spars. The difference lies in the type of the landing gear. Since the landing gear is mounted on the wing, straight spars must be installed, similarly to the Space Shuttle. The wing model in CATIA (a CAD design software) is presented on Figure 14. The main spar, in the middle of the wing, is sized with numerical simulations to bear half the weight of the vehicle in the worst case and ribs are placed every 500 mm.

3. Fuselage study

The number of frames carrying the wing is calculated based on the load factor. Indeed, since the wings bear the weight of the vehicle along all its length, a significant torque is present at each wing root. It is assumed that this force is applied at the mean aerodynamic chord. Even if according to the calculations, a single main frame with a thickness of 100 mm is able to carry the entire structure, another one is added for safety. The other frames act as secondary structures between compartments and avoid buckling. Another frame is sized to handle the compression/traction of the longerons which propagates the efforts of the main engine. The spars are sized to withstand the thrust of the hybrid engine. Finally, nine spars are identified as a good compromise to carry the loads without being too heavy. At the engine interface, titanium should be used due to strong constraints. A simple representation of the fuselage with only its spars and frames is modeled on Figure 15.
K. Reentry and thermal protection system design

1. Trajectory

Once the rocket engines are shut off, the vehicle is only subject to its weight and follows a parabolic trajectory to reach the targeted altitude before reentering the atmosphere. A modeling and simulation environment is created to model the trajectory, the thermal constraints and accelerations during this critical reentry phase.

Equations 22 and 23 come from N. Vinh et al.\textsuperscript{20} who apply Newton’s Second Law to the reentry body. \( V \) is the speed, \( \gamma \) the flight path angle, \( C_l \) and \( C_d \) the lift and drag coefficients, \( m \) the mass of the vehicle, \( r \) the distance of the vehicle from the Earth center and \( \rho \) the density. M. Grant and R. Braun\textsuperscript{21} provide analytic formulas for the hypersonic aerodynamic coefficients based on a simplified geometry.

\[
m\frac{dV}{dt} = -\frac{1}{2} \rho SV^2 C_d - mg_0 \sin \gamma \tag{22}
\]

\[
mV \frac{d\gamma}{dt} = \frac{1}{2} \rho SV^2 C_l - m \left[ g_0 - \frac{V^2}{r} \right] \cos \gamma \tag{23}
\]

Regarding the specifications of the vehicle, the main constraint on the trajectory is the maximum allowed acceleration \( n_{\text{max}} \) (fixed at 3 g for the reentry). This constraint is used to build a control strategy during reentry. Equation 24 mathematically describes the constraints that drive the optimization process. The angle of attack \( \alpha \) is controlled so that the acceleration felt by the passengers \( n \) is as close as possible to the maximum while remaining below the maximum allowed angle of attack \( \alpha_{\text{stall}} \). The reentry is assumed to end when the flight path angle is low enough.

\[
\begin{align*}
n &\leq n_{\text{max}} \\
\alpha &\leq \alpha_{\text{stall}} \\
0 &= (\alpha - \alpha_{\text{stall}}) \times (n - n_{\text{max}})
\end{align*} \tag{24}
\]

Figure 16 shows the acceleration felt by the passengers during reentry. The vehicle starts a typical gliding descent at an altitude of 11 km, with a speed of 120 m/s to match the targeted descent path. The computed trajectory satisfies the maximum load factor constraint. A less constraining load factor limit would decrease the thermal load but deteriorate the comfort of the passengers. Based on these mechanical constraints, the thermal protection system and the structure of the vehicle can be designed.
2. Thermal protection system

The mechanical loads undergone during reentry are the main design drivers for the thermal protections. The latter are particularly important and have already caused crew losses. This section is based on the methodology given by N. Vinh et al.\textsuperscript{20} The main concepts to study are the total heat flow received by the vehicle during the mission, the heat rate, the dynamic pressure and the thermal gradients. The first one is important because of the limit to the quantity of heat that can be absorbed by the vehicle.

The heat mainly comes from the deceleration that is transformed into heat around the vehicle. Then, this heat is whether transmitted to the vehicle or to the atmosphere through a shock wave. The goal is to minimize the part absorbed by creating the largest shock wave possible. The total incoming convective flux $Q_{\text{conv}}$ is given by Equation 25. $V_0$ is the speed at the beginning of the reentry, $C_f$ the friction coefficient, $m$ the mass of the vehicle and $A$ the exposed surface. According to N. Vinh et al.,\textsuperscript{20} the ratio $C_f/C_d$ peaks at 0.1 for a lifted reentry.

\[ Q_{\text{conv}} = \frac{1}{2}mV_0^2 \times \frac{1}{2}A \frac{C_f}{5 \times C_d} \]  

Equation 26 provides the incoming heat rate by convection. For a lifted reentry, the $C_f/C_d$ ratio is higher compared to a ballistic one, hence the total heat flux is also higher. To mitigate this effect, radiative surfaces are used so that they can radiate heat at a rate up to the maximum incoming rate.\textsuperscript{20} Moreover, a high angle of attack is present at the beginning of the reentry to decrease the total incoming heat. The angle of attack is then reduced over time to decrease the drag coefficient.

\[ q_{av} = \frac{1}{4} \rho V^3 C_f \]  

During the reentry, the proportion of heat transmitted to the vehicle by convection is around 0.15. The maximal flux obtained is approximately 0.6 MW/m\textsuperscript{2}. According to R. East,\textsuperscript{21} trans-atmospheric vehicles such as Hotol have a maximal heat flux of around 0.5 MW/m\textsuperscript{2} while the one of the Space Shuttle was 20 MW/m\textsuperscript{2}. These values confirm the order of magnitude of the calculations. Finally, even if the nose and the leading edges must be protected more, the thermal protection system is lighter than the one of the Space Shuttle since it does not require tiles.

This section described the sizing method for each subsystem. Detailed results were given for the most important components such as the wing and the propulsion systems. Each piece has been integrated into Matlab in order to benefit from a single environment that has been run until convergence on all values. The latter describe a new concept whose true performance must now be studied in order to fully validate the initial requirements.

IV. Performance Study and Requirements Validation

The aforementioned converged values provide sufficient information to model the entire vehicle into CATIA. Moreover, it also enables a more accurate description of the aerodynamic of the vehicle and its propulsive performance. Thus, a geometric description and a mass budget will be presented before simulating the complete mission.
A. Vehicle characteristics

Figure 17 displays the 3-view drawing, modeled in CATIA, of the vehicle and shows the major dimensions (in mm) of the main parts of the vehicle: fuselage, wing, empennage, turbofans and hybrid engine nozzles.

The nose of the vehicle mainly stores the attitude control thrusters. The cockpit is 2.03 meter long and contains the navigation-related avionics, the nose landing gear, the two pilots’ seats and their life module. A fire-wall separates the cockpit from the cabin which is 4.5 meter long, containing eight seats as well as the environmental control system. A 50 cm wide zone stores some subsystems such as hydraulic and electric components. Then, the remaining part of the fuselage is divided between the fuel tanks for the turbofans (0.6 meter long), and the fuel tanks for the hybrid engines which are 8.8 meter long (for the tanks and the combustion chambers). It ends with the nozzles (1 meter long), which are protected from thermal charges with a fuselage flap. Tables 5, 6 and 7 sum up the dimensions of the vehicle.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Span</td>
<td>10.04 m</td>
</tr>
<tr>
<td>Root chord</td>
<td>9.6 m</td>
</tr>
<tr>
<td>Tip chord</td>
<td>1.3 m</td>
</tr>
<tr>
<td>Inner sweep angle</td>
<td>81°</td>
</tr>
<tr>
<td>Outer sweep angle</td>
<td>45°</td>
</tr>
</tbody>
</table>

Table 5. Wing dimensions

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter</td>
<td>1.9 m</td>
</tr>
<tr>
<td>Cockpit length</td>
<td>2.03 m</td>
</tr>
<tr>
<td>Cabin length</td>
<td>4.5 m</td>
</tr>
<tr>
<td>Subsystems zone length</td>
<td>0.5 m</td>
</tr>
<tr>
<td>Turb风扇 fuel tank length</td>
<td>0.6 m</td>
</tr>
<tr>
<td>Hybrid engine length</td>
<td>8.8 m</td>
</tr>
<tr>
<td>Nozzles length</td>
<td>1 m</td>
</tr>
</tbody>
</table>

Table 6. Fuselage dimensions

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Span</td>
<td>2.431 m</td>
</tr>
<tr>
<td>Root chord</td>
<td>4.120 m</td>
</tr>
<tr>
<td>Tip chord</td>
<td>2.03 m</td>
</tr>
<tr>
<td>Sweep angle</td>
<td>40°</td>
</tr>
</tbody>
</table>

Table 7. Vertical stabilizer dimensions
B. Empty weight estimation

The mass budget of the vehicle is based on statistical models taking into account civil, military and hypersonic aircraft as well as the Space Shuttle. Only the propulsion part was calculated from the studies carried out. Several models were developed in the literature and are compared by R. R. Rohrschneider.\textsuperscript{23} The differences between the models have been assessed and, in most cases, average values are used. The references of the models are NASA’s Technical Memorandum 78661\textsuperscript{24} and T. Talay’s class notes.\textsuperscript{25} 20\% is added as a design margin, but in some parts the masses are reduced to account for technological enhancements, especially in structure. The masses of the different elements of the fuselage are provided in Table 8.

Raymer’s methodology\textsuperscript{10} is used to compute the mass of the landing gears. While the main landing gear weighs 241 kg, the nose landing gear only weighs 40 kg. Since the reentry of the vehicle would be much lighter in terms of load factor than the Space Shuttle’s, mass reductions are taken into account. The formula used to calculate the mass comes from T. Talay.\textsuperscript{25} A mass of 290 kg is obtained and 150 kg are allocated to multiple interfaces. The hybrid engine has an empty mass of 2,539 kg with 11,892 kg of fuel. Each turbofan has a mass of 800 kg. 100 kg is added for the air induction system in each wing, as well as 10\% of the total mass for the attachment of the engines. 2.07 tons are therefore allocated to these engines, and their fuel weighs 2.3 tons. The attitude control system has a structural mass of 315 kg and a fuel mass of 26 kg. Ten people of 100 kg are considered, including two pilots. 200 kg is allocated to the life support system. The equipment compartment contains instruments for navigation, electricity and miscellaneous. The avionics mass is estimated from Raymer.\textsuperscript{10} The flight control system mass comes from A. Wilhite.\textsuperscript{26} Finally, the total mass allocation is given in Table 9 and Figure 18 shows the distribution of mass within the vehicle.

\begin{table}[h]
\centering
\begin{tabular}{|l|c|}
\hline
Sub-structure & Mass \\
\hline
Outside structure & 1,729 kg \\
Secondary structure & 107 kg \\
Cockpit & 1,206 kg \\
Fuselage flap & 25 kg \\
Hybrid engine’s attachment & 81 kg \\
Rudder & 107 kg \\
\hline
Total mass fuselage & 3,149 kg \\
\hline
\end{tabular}
\caption{Masses of the different parts of the fuselage}
\end{table}

\begin{table}[h]
\centering
\begin{tabular}{|l|c|}
\hline
Subsystem & Mass \\
\hline
Structure & 5,445 kg \\
Propulsion & 4,834 kg \\
Fuel & 14,218 kg \\
People and payload & 1,300 kg \\
Equipment & 1,693 kg \\
\hline
Empty mass & 13,271 kg \\
Take-off mass & 27,489 kg \\
\hline
\end{tabular}
\caption{Mass budget}
\end{table}

C. Mission description

The result of the design process is a 16.43 meters long suborbital vehicle with a span of 10 meters able to carry 8 passengers and 2 pilots, with a total weight of 27.5 tons. In order to analyze the performance of the vehicle, the mission is divided into several segments. It starts with a warm-up phase of 3 minutes. Then the decision speed during the take-off phase calculated with the standards from the Federal Aviation Regulations (FAR 25.225) is \( V_1 = 88 \text{ m/s.} \) The take-off field length is 2,070 m. Once the take-off is finished, the vehicle accelerates up to an initial climbing speed of 121 m/s, at the optimum rate of climb. The jet-powered climb is performed at the optimum rate of climb up to an altitude of 11 km. This first part of the mission ends at a velocity of 292 m/s with 756 kg of fuel consumed. Then, the turbofans are shut down and the hybrid engine is ignited to perform the rocket climbing phase. 11.5 tons of propellant are consumed during 78.6 s. The vehicle reaches the altitude of 45.6 km and the maximum load factor is 3.26 g when the hybrid engine is cut off with a Mach number of around 3. Then the weightlessness phase starts and lasts 3 min 46 s with 1 min 26 s spent in space (above 100 km). The vehicle reaches a maximum altitude of 109 km. The reentry phase is a lifted reentry which lasts around 1 min 30 s, and ends when the vehicle comes back to an
altitude of 11 km with a velocity of 120 m/s. The vehicle performs a gliding descent until an altitude of 5 km, where the turbofans are re-ignited for a safe approach and landing. The approach speed is 73 m/s and the landing distance is 864 m. The mission ends with a taxi-out phase of 3 min. After the parabola phase, the fuel consumption for the approach and landing is 174 kg. A classic safety margin of 15 min time of flight is taken, which adds 1,322 kg of fuel supply. Figure 19 shows the trajectory of the mission.

Figure 19. Trajectory of the vehicle during its mission

A risk analysis of each flight phase is also conducted in order to identify and mitigate the risks that can be encountered in terms of environment, psychology, vehicle failures and other events. Several levels of consequences are defined to compare them: failure not leading to the failure of the mission, failure of the mission (mission aborted but no damage), damages on the vehicle, injured crew, loss of the crew. The goal is to decrease the risk in order to match the loss probability of commercial aircraft flights. First, the use of hybrid propulsion decreases the risk of catastrophic failure compared to the other types of propulsion. The reentry is the most critical part of the mission with a higher probability of failure, and it affects the landing phase. Indeed, if problems occur during this phase, they might deteriorate the landing capabilities of the vehicle. In space, the main risks are the micro-meteoroids which have been included in the design constraints for the structure.

V. Conclusion

Based on an emerging demand for space tourism, this work demonstrates the need of a new suborbital vehicle with larger and safer characteristics. A typical aircraft sizing and synthesis methodology was modified to match specific suborbital vehicle requirements such as the rocket climb and reentry phases. A first conceptual design phase provided the main characteristics of the vehicle in terms of thrust, wing area and weight. Then, a more detailed study of the different subsystems allowed the calculation of the dimensions and parameters of the propulsion systems, the cabin layout, the wing shape, the vertical tail, the landing gear and the cockpit. The structure design was finalized by studying the two most constraining parts of the mission: the rocket climbing phase and the reentry with its thermal loads. Once each subsystem has been designed and implemented within the overall vehicle with CATIA, a complete geometric description combined with a detailed mass budget provide an accurate simulation of the mission. The latter is finally used to validate the requirements. Indeed, the designed vehicle is able to carry eight paying passengers (more than most of the existing concepts). Moreover, in addition to the two pilots, the double propulsion system also improves safety: hybrid rocket engines are safer than liquid or solid propulsion systems and the turbofans can be restarted for a powered landing. In terms of comfort, the maximum load factor has been reduced to 3.3 g. Concerning the passenger experience, the weightlessness phase lasts 3 min 26 s including 1 min 26 s spent in space. During the parabola, three turns are performed to allow all passengers to see the curvature of Earth by reaching an altitude up to 109 km.

Some companies which are planning to enter the manned suborbital market such as Swiss Space Systems could benefit from this work. Indeed, an airborne launched suborbital vehicle is only appropriate for small payload trans-
portation such as small satellites because the size of the vehicle is limited by the size of the carrier. Thus, another concept would be needed to carry heavier payloads and especially a growing number of passengers. Consequently, this work would be a good starting point for a safer concept that would match the emerging touristic demand. Nevertheless, more precise models could be developed and CFD (Computational Fluid Dynamics) calculations would improve the accuracy of the performance calculation. A business case should also be performed in order to check the economic viability of the vehicle.

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