Preliminary Sizing of an Hypersonic Airbreathing Airliner

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The purpose of this paper is to identify, for given technology levels (TRL) and mission requirements, those parameters that are critical for preliminary sizing of a hypersonic airbreathing airliner. Mission requirements will dictate a solution space of possible vehicle architecture capable of meeting cruise conditions as well as of taking-off (TO) and landing. In practice, once defined a range of cruise vehicle architectures, constraints are imposed (as to all passenger airliners), such as: 1. take off (=TO) and landing distance (so-called field length, FL): FL no longer than for the B-747-400, or 10000 ft; 2. completing TO with one engine off; 3. max acceleration at TO and climb-out (CO) = 0.4 g; 4. Hydrogen fuel (Meeting NOx emission limits (EINOx) is a further constraint not discussed in this paper).

These constraints enable focusing on a realistic design out of the broad range of vehicles capable of performing the given mission. Thus a realistic vehicle must not only integrate aerodynamics and propulsion system; in fact, it is the result of many iterations in the design space, until performance and constraints are successfully achieved and met.

The Gross Weight at Take Off (TOGW) was deliberately discarded as a constraint, based on Previous studies by Czysz. Typically, limiting from the beginning the TOGW leads to a vicious spiral where weight and propulsion system requirements keep growing, eventually denying convergence. In designing passenger airliners, in fact, it is the payload that is assumed fixed from the start, not the total weight.

A parametric analysis of the hypersonic vehicle architecture is presented: in particular, optimal size, weight and geometrical shape are defined for different mission requirements. This analysis has shown that, it is possible to define a range of possible successful solutions for the European LAPCAT II project.

Key Words: Hypersonic Vehicle, Scramjet Sizing, Küchemann \(\tau\)

Nomenclature

- \(\Delta V\): speed increase
- \(\eta_v\): useable Volume
- \(\theta\): propulsion energy conversion efficiency
- \(\tau\): Küchemann \(\tau\) parameter
- \(a\): sound speed
- \(ff\): Fuel fraction
- HTOL: horizontal Take-off and landing
- \(K_w(\tau)\): wetted-to-planform area ratio parameter
- \(Isp\): Specific Impulse
- \(I_{str}\): structural index
- \(L/D\): Lift over Drag
- \(M\): Mach number
- OWE: Overall empty weight
- Qcc: combustion heat
- \(r_{sys}\): System weight to gross weight
- RF: Range factor
- \(S_{plan}\): Planform Surface
- \(S_{wet}\): Wetted Surface
- \(T/W\): engine Trust to Weight ratio
- TO: Take-off
- TOGW: Take-off Gross Weight
- \(V_{fuel}\): Fuel Volume
- \(V_{pay}\): payload Volume
- \(V_{tot}\): total Volume
- \(V_{void}\): void volume
- \(W_{fuel}\): fuel weight
- \(W_{pay}\): Payload weight
- \(W_{prop}\): propulsion weight

1. Introduction

The current interest in hypersonic airbreathing vehicles for new generation high-speed commercial airplane, launchers and trans-atmospheric vehicles requires a better understanding of the critical parameters to size hypersonic airbreathing vehicles. Studies on hypersonic configurations dated back to
early sixties in USA, Russia and EU. Currently, the European project LAPCAT-2 has the ambitious goal to define a conceptual vehicle able to achieve the anti-podal range Brussels-Sydney (~18000 km) in about 2 hours at Mach 8. At this high speed, the requirement of high L/D is critical to achieve because of the high skin friction drag and high wave drag: in fact, L/D decreases as the Mach number increases. The design of vehicle architecture is crucial to meet the high L/D requirements.

Previous studies by Czysz\textsuperscript{1,2} have shown that the approach of integrating individually optimized system elements across matching yielded a significant reduction in performance. In fact, in supersonic aircraft each component was independently sized, designed and assembled, in particular the design of the vehicle began by drawing constant wing area or constant weight concept aircraft. Actually, as the speed increases the controls configuration becomes increasingly critical. From Mach 4.5 to Mach 6.0, the pitch and roll control move from trailing edge devices to all flying surfaces end-plated by the vertical tails. The total area of the tip control surfaces can be as much as 14 percent of the total platform area. In the new approach, introduced by P. A. Czysz, the sizing of the vehicle begins from the mission distance and maneuver performance requirements. In this new approach, by looking at the range and at the Mach number it is possible to define a set of configurations that allows the convergence of volumes and weights to be obtained for the mission requirements.

The procedure to size the vehicle is based on a strategy maximizing the so-called ‘available energy’ and originally developed many years ago\textsuperscript{3}. In this concept the entire vehicle is considered a thermal machine, where, in general, heat is not only supplied by combustion of hydrogen and air, but also consists of heat fluxes entering the vehicle surface due to convective HT (heat transfer) and friction work. The useful work is that of the thrust applied to the vehicle flying at its instantaneous speed. In essence, and if the thermodynamic sink is the outside atmosphere, this coincides with the work available as the thermodynamics free energy concept, where irreversible losses limit the amount of work that can be extracted from heat addition. In this sense this strategy includes minimizing entropy production due to irreversible losses, and has been rediscovered under various names (e.g., exergy analysis). An example is in\textsuperscript{4} and includes both first and second principles of thermodynamics in their mechanistic formulations.

In this context, to reduce irreversible losses due to vehicle aerodynamics, the work of Küchemann\textsuperscript{5} to maximize L/D has been embodied in the form of the all-important parameter $\tau$:

$$\tau = \frac{V_{\text{ref}}}{S_{\text{plan}}}$$  \hspace{1cm} (1)

defined as a sort of ratio between volume divided by surface area\textsuperscript{3,2}.

At this stage of our knowledge it is in principle feasible to write down all the relevant equations, whether first principles or semi-empirical correlations, and solve them at once. However, the non-linear nature of all equations, as seen later, produces a set of equations where convergence to a physically realistic solution is ensured only if the initial guess is reasonable. This is in fact hard to obtain by the very nature of the problem; thus it is expedient and far more convenient, to proceed by iteratively comparing volume and mass budgets obtained by partial solutions, as explained in the next Sections.

The work and results described here focus on the key aspects of the vehicle, seen as a single integrated unit, stressing reliability and consistency of the whole rather than emphasizing rigorous analysis of individual ‘blocks’. If a lesson can be drawn from the past, it is that hypersonic vehicles should be designed as a single unit, i.e., and not as an assembly of even individually optimized components.

2. Hypersonic Vehicle Design Background

A lesson learned in the past\textsuperscript{3} is that hypersonic vehicle sizing is very different from that for subsonic and supersonic aircraft. In fact, in hypersonic vehicles sizing begins from the mission distance and payload and not by drawing constant wing area or constant TOGW aircraft. Significant differences between conventional and hypersonic aircraft are the huge propellant volume and the low aerodynamic efficiency L/D. The NASA AMI-X program showed that the approach of integrating individually optimized system elements (as done in subsonic and supersonic aircraft) reduced performance significantly: the sum of individually optimized subsystems does not result in a system optimum. A system optimum is the sum of subsystems that yields the optimum system, not a collection of optimum components.

![Integration through the total system performance maximization](image)

Fig. 1. Integration through the total system performance maximization [Czysz, 2000].

Thus, the starting point for hypersonic vehicle design is a first order analysis of a purposely-defined vehicle figure of merit, instead of the single optimization of different components. In fact, at high Mach numbers, high L/D ratios are very difficult to achieve because of high skin friction and wave drag: hypersonic L/D are of order of 2-8, far below those of conventional aircraft (see Fig. 2). For example, maximum L/D \textasciitilde 12 to 15 are common for designs like the P-51 Mustang fighter and F-111 bomber. Küchemann analyzed
this trend and formulated the general empirical relationship:

\[
\frac{L}{D}_{\text{max}} = \frac{4M_\infty + 3}{M_\infty}
\]  

(2)

This relationship, based on actual data from flight vehicles and wind tunnel results, has proven generally accurate across the supersonic and hypersonic regimes. Küchemann’s equation (solid line in Fig. 2) is plotted together with a series of actual results for some hypersonic vehicles (open circles).

In fact, such system is so integrated that it is difficult to isolate individual elements in terms of traditional technical disciplines. That is, the propulsion configured vehicle underside does produce thrust, while it adds about a 50% increment to the aerodynamic lift at hypersonic speeds, and increasing the lift-to-drag ratio from 4.0 to 6.0\(^{10}\). Manipulation of the combustor module thrust vector has six times the pitching moment effectiveness of the aerodynamic tip controls with 1/100 the area (these figures are from actual wind tunnel test data). The net result is that the L/D of the total system is better than the ideal aerodynamic shape, in spite of the fact that the performance of each of the individual components is lower at the design Mach number than that of the previously optimized components. The reason is that aerodynamic compression due to the forebody is not re-expanded to free-stream speed around the fuselage and, as an extreme example, at Mach 12 flight conditions, an engine cowl operates at Mach 6 rather than Mach 12. The resulting reduction in the entropy rise means that there is more available energy that can be converted into thrust, and provides a portion of the improved combined performance.

So, a convergence solution space that links the vehicle mission requirements to aerodynamic and propulsive efficiencies is the answer to the challenges of hypersonic vehicle design.

Past experience concluded that for an airbreathing concept operating at \(M>6\), the family of geometric configurations has to be one of the following three: blended body, wing body or waverider, because these are designed at the underside of vehicle, via a being multishock inlet system, to provide air to the engine cowl face.

For these three families, and using hydrogen fuel, the minimum gross weight and minimum empty weight generally occur in the vicinity of the Küchemann \(\tau=0.18\). For HTOL the minimum sized aircraft (\(\tau=0.18\)) has too little planform area: this must be increased until the HTOL constraints are met. This means increase the planform area, i.e., decrease \(\tau\) until the minimum dry weight is reached. Thus, what is expected for a hydrogen fuelled aircraft (as found in past studies), is a solution in the range of \(\tau = 0.14\) to 0.18\(^{12, 10}\).

3. Approach

The approach for hypersonic vehicle sizing just sketched is in practice based on what will be called here the VDK (Vanderkerkhove)/HC (Hypersonic Convergence)\(^3\) parametric sizing methodology. This methodology was developed since the ‘80s and applied to:

(a) high-performance subsonic to hypersonic aircraft
(b) reusable space launchers

In this approach, components are NOT independently sized, designed and assembled; sizing begins with the mission distance, payload and cruise Mach number, to obtain a figure of merit (\(\tau\)) of the whole vehicle. The VDK
sizing methodology is based on the simultaneously solution for the OWE and $S_{plan}$ equations, ensuring that the separately calculated available and required weights and volumes converge for a given Küchemann’s τ parameter. To explain, note that all variables in these calculations are strictly connected to each other. For example, if the range increases, the propellant weight also increases. The increase of the propellant weight drives the weight increase of all systems and of the structure. The same occurs for the propellant volume: increasing its volume drag increases and a larger planform area is needed to produce higher lift and keep L/D reasonable. Larger planform area means more wetted area, the structural weight increases too, and the larger TO gross weight requires more propellant. Thus this process may diverge, and that is why a solution must be found by solving simultaneously the set of equations that relate all dependent variables (volumes, weights and vehicle geometry) to the mission input (M, L/D, range, and payload). Since these equations are nonlinear, they must be iterated until, for instance, the volume required (from the desired performance and constraints) is equal to the volume available (from aerodynamics and structure). The same holds in terms of weight.

To identify operational and technical lessons learnt in the past, a survey of representative high speed aircraft studies has been carried out first. Four configurations have been chosen as reference in the present work:

1. HYCAT-1A
2. Sanger II
3. ELAC
4. DF-9

Table 1. Representative Past High Speed Aircraft Studies –Survey

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Research Group</th>
<th>Lead Investigator</th>
<th>Year</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hypersonic Cruise Aircraft (HYCAT)</td>
<td>Lockheed-California Company</td>
<td>R. E. Morris/G. D. Brewer</td>
<td>1979</td>
</tr>
<tr>
<td>Sanger II</td>
<td>MBB</td>
<td>D. E. Koelle</td>
<td>1987</td>
</tr>
<tr>
<td>Elliptical Aerodynamic Configuration (ELAC)</td>
<td>RWTH Aachen</td>
<td>D. Jacob</td>
<td>Approx. 1993</td>
</tr>
</tbody>
</table>

These configurations refer to wing body, blended body, elliptical cone body, and a waverider. They have been chosen to generate reference values to calibrate and validate the sizing results using this approach.

The Küchemann τ of these configurations typically ranges from 0.06 for a blended body/waverider, to 0.218 for a wing body. For a given mission requirements more than one configuration was found, and it is the constraints of mission typology (commercial aircraft, space plane, launcher…) that will define the “best configuration”.

4. Implementation Procedure at the Cruise Design Point

In this initial analysis, a solution space of aircraft configurations for given design specifications has been defined by solving simultaneously all “cruise” equations. Actually, once defined a range of possible solutions at the cruise design point, the overall mission, i.e., from TO to landing, must be accounted for. The procedure is:

1. define known and unknown variables
2. implement equations
3. solve equations iteratively until weight and volume converge.

4.1 Definition of known and unknown variables

A set of known and unknown variables define the present analysis. The known variables are mission requirements, performance, and industrial technology level.

In the present work, mission requirements are those in the EU-funded LAPCAT II project:

1. Cruise Ma=8
2. Range=18,728 km
3. N of pax=300 ($W_{pay}=60$ ton)
4. hydrogen as fuel

At cruise, the performance assumed is:

5. $Isp=2000$ s
6. $T/W_{sys}=8.3$

These performance parameters have been set only to define a preliminary figure of merit and vehicle weights and volumes, thus are not final. This done, a trajectory is defined: along this trajectory, fuel consumption, weights, and actual $Isp$ and L/D must be re-calculated and implemented as starting point to calculate all variables for the whole mission, from TO to landing. Once the new values are calculated, the trajectory is again calculated, in order to fit the input data and used again as the starting point to calculate the design specifications: this is repeated until design specifications fit the mission trajectory.

Industrial Technology variables have also been fixed:

7. $Istr$ (structural index)=15, 18, 21 kg/m
8. $W_{sys}/W=0.07$
9. $\eta_s=0.7$ (useable Volume)

The $Istr$ defines the level of maturity and technology of structural materials: the lower the $Istr$, the more sophisticated the structure.

Summarizing what said, the known/given variables are 9. The unknowns are 14: $\tau$, $S_{plan}$, $S_{wet}$, $V_{fuel}$, $V_{sys}$, $V_{prop}$, $V_{liferef}$, $V_{lifetot}$, $\eta_s$, $\eta_f$, $\eta_v$, $L/D=f(Ma, \tau)$, TOGW, $W_{sys}$, $W_{prop}$, $W_{fuel}$, $W_{stb}$, $K_{sys}(\tau)$.

In order to determine the unknowns, a set of equations has been defined and then solved.

4.2 Definition of equations

In order to close this problem mathematically, a set of equations relating the vehicle architecture to the vehicle performance and mission requirements (by means of the Küchemann τ) must be written.

The first equation is given by the Küchemann τ parameter
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A definition linking the vehicle total volume, \( V_{tot} \), to the planform area, \( S_{plan} \), (see Eq. (1)).

The Küchemann’s \( \tau \) varies from \( \tau = 0 \) for a sheet and \( \tau = 0.75225 \) for a sphere.

The Küchemann’s \( W \) varies from \( W = 0 \) for a sheet and \( W = 0.75225 \) for a sphere.

Eq. (1) shows that as \( \tau \) increases the Planform Surface decreases and available volume increases.

For the same planform area, as the volume increases, the aerodynamic drag increases too: in order to have ‘good’ L/D a small \( \tau \) should be chosen (see Fig. 3). The infinitely thin flat plate (\( \tau = 0 \)) represents, in fact, the most efficient hypersonic lifting configuration in terms of L/D. However, it has no volume for payload, engines, fuel, etc. So, an efficient vehicle architecture depends on the minimum volume required to contain fuel, passengers, engines, systems... and at the same time on the minimum planform area needed to ensure a L/D high enough to meet mission requirements.

Increasing the planform area, for a given \( \tau \), increases structural weight and (among other things) cost.

The equation:

\[
S_{wet} = K_w(\tau) \times S_{plan} \tag{3}
\]

relates the planform area to the wetted area by means of \( K_w \), a function of \( \tau \) known from structural and aerodynamic experience.

Fig. 3. General trends with varying \( \tau \).

Fig. 4 shows that \( K_w \), and thus the wetted-to-planform area ratio, increases with \( \tau \). Different relationships hold for different geometry configurations.

Increasing \( \tau \) increases the wetted area, and as a consequence friction drag. For a given total volume, as \( \tau \) increases (say, going from a waverider, with minimum \( \tau \), to a sphere, with the highest \( \tau \)) the wetted surface decreases (see Fig. 5). This is due to the fact that, for a given total volume, if \( \tau \) increases the planform area decreases more than the increase of the wetted area. For a given wetted area, increasing \( \tau \) raises the volume while L/D decreases. The choice of vehicle configuration depends on the volume required by the fuel, engine, systems and the payload, and on the L/D needed by the range and cruise Mach number.

Fig. 4. \( K_w \) vs \( \tau \) for different vehicle configurations.

From flight and wind tunnel data, this L/D is a function of \( \tau \) and of the Mach number:

\[
L/D = \frac{A(M + B)}{M} \left( \frac{1.0128 - 0.2797 \ln(\tau/0.03)}{1 - M^2/673} \right) \tag{4}
\]

where \( A = 6 \) and \( B = 2 \).

Fig. 6 shows L/D as function of M and \( \tau \). L/D decreases with increasing Mach number and increases as \( \tau \) decreases.

The fuel volume:

\[
V_{fuel} = \frac{W_{fuel}}{\rho_{fuel}} \tag{5}
\]

depends on vehicle performance L/D(\( M, \tau \)), Isp(\( M \)) and the mission range.

In fact, the fundamental Bréguet Range Equation links range, fuel fraction, fuel combustion energy to aerodynamic and propulsion efficiency.

For cruise conditions:

\[
Range = -RF \times \ln(1 - ff) \tag{6}
\]

where the range factor, RF, is:
and

\[ \text{ff} = \frac{W_{\text{fuel}}}{W_{\text{TOGW}}} \]  

(8)

The Breguet range equation (see Fig. 7) shows that the range is primarily a function of:
1. aircraft configuration
2. characteristic of the propulsion concept and of the propellant
3. L/D

The Range is also a function of the Mach number through Isp and L/D. For a given Isp and L/D, the range factor increases with Mach number.

\[ \text{Range} = L \times \text{Isp} \times \text{fuel} \times \frac{\Delta V}{\text{D}} \times \frac{\text{M} \times \text{Isp} \times \frac{\text{L}}{\text{D}}}{\text{D}} \]  

(7)

Fig. 6. L/D vs M with \( \tau \) as parameter.

The payload volume:
\[ V_{\text{pay}} = \frac{W_{\text{pay}}}{\rho_{\text{pay}}} \]  

(10)

depends on mission typology. The payload weight has been (pessimistically) estimated about 60 ton. This value is consistent with (18-21). For example, the Concorde passengers weight was assumed to be 100 kg per passenger, that means 30 ton for 300 passengers plus 30 ton for seats, toilets, air conditioning and all safety systems.

The payload density for a commercial aircraft is typically
70-120 kg/m³. 100 kg/m³ is assumed in this work.

The void volume:

\[ V_{\text{void}} = V_{\text{tot}} \times (1 - \eta_v) \]  \hspace{1cm} (11)

has been estimated assuming \( \eta_v = 0.7 \).

The total volume available, calculated from \( \tau \), has been iterated until converged with the volume required to accommodate the payload, the fuel, and all systems.

\[ V_{\text{tot}} = V_{\text{pay}} + V_{\text{fuel}} + V_{\text{void}} \]  \hspace{1cm} (12)

The weight of all systems is given from:

\[ W_{\text{sys}} = \frac{W_{\text{TOGW}}}{\text{ETW}} \left( \frac{1}{L/D} \right) \]  \hspace{1cm} (13)

where \( \text{ETW} \) is the engine thrust-to-weight ratio, assumed 8.3.

The structural weight depends on the wetted surface and the structural weight index:

\[ W_{\text{str}} = \frac{W_{\text{swet}}}{\text{S}_{\text{plan}}} \times \text{S}_{\text{plan}} = \text{Istr} \times K_w \times S_{\text{plan}} \]  \hspace{1cm} (15)

The lower the structural weight index, the more advanced technologically the solution required. Three Ist have been assumed to calculate structural weight: 15, 18, and 21 kg/m². Once \( W_{\text{pay}}, W_{\text{fuel}}, W_{\text{sys}} \) and \( W_{\text{prop}} \) are calculated, the TOGW can be calculated:

\[ \text{TOGW} = W_{\text{pay}} + W_{\text{fuel}} + W_{\text{sys}} + W_{\text{prop}} \]  \hspace{1cm} (16)

This weight has been iterated until converging with that calculated from the minimum volume requirement equations. The entire set of equations must be simultaneously solved to find a solution for the volumes and weights at convergence. The number of variables is 23, and 9 are given; the number of equations is 13: therefore there are \( \infty^P \) solutions. This means that for the same \( M \), range and payload requirements, a solution space exist rather than one (unique) solution.

The solution of choice results from constraints that depend on mission typology (e.g., commercial aircraft, military aircraft, launcher…).

For a commercial aircraft, passenger safety and comfort (acceleration) constraints are key requirements, as well as airport constraints.

5. Space Solution of Hypersonic Aircraft

Above \( M = 6 \), five reference configurations are considered as possible choices: wing body, blended, elliptical cone, half elliptical cone and waverider.

Converged solutions for these different configurations have been calculated. Here only the solution space for a wing-body is reported.

The solution space for a wing-body, as a function of \( W_{\text{str}} \) and \( S_{\text{plan}} \), for the total volume available (calculated from \( \tau \)), and converging to the volume required and calculated from TOGW is shown in Fig. 11.

5.1 Wing body configuration analysis

Fig. 12 (the projection of \( V_{\text{tot}} \) vs \( S_{\text{plan}} \) of Fig. 11) shows a curve locus of solutions for the same mission requirements. The \( \tau \) and Splan values fall within the nominal limits defined. In fact, with hydrogen fuel, the region is \( 0.14 \leq \tau \leq 0.21 \). Different design specifications (\( S_{\text{swet}}, V_{\text{swet}}, V_{\text{void}}, V_{\text{fuel}}, L/D, \text{TOGW}, W_{\text{sys}}, W_{\text{prop}}, W_{\text{fuel}}, W_{\text{str}}, K_w(\tau) \)) correspond to these \( \tau \) values. The solution curve has two minima, a minimum planform area and a minimum wetted surface: \( \tau = 0.16 \) corresponds to a solution with a minimum \( S_{\text{plan}} \) while \( \tau = 0.14 \) correspond to a minimum \( S_{\text{swet}} \).
Fig. 12. Solution space identified by sizing process defines planform area and total volume.

These solutions are very close together, from 3000 m$^2$ at 970 m$^2$ to 2900 m$^2$ at 1000 m$^2$, but there is still a range of solutions to choose from.

A minimum $S_{sw}$ means a minimum structural weight, i.e. 52 ton at $\tau=0.14$ and 2 ton extra at $\tau=0.16$.

Fig. 13. Swet vs Splan of convergence solutions.

Fig. 14. Structural weight is primarily a function of aircraft size.

The structural weight, $W_{str}$, has two closely spaced minima that are very reasonable a high skin temperature cruise (when the mass of the TPS is significant). The Operational Empty Weight (OEW) varies by 7 ton between these two minima. Empty Weight spans between a minimum 82 ton for a 1000 m$^2$ planform area and 88 ton for a minimum planform area 970 m$^2$. The OWE, Operational Weight Empty (that is OEW plus payload), is about 190 ton, for a mass ratio=3.

This OWE shows a minimum 188 ton for $\tau=0.15$. A typical OWE range is between 188 ton to 190 ton for the two minima: the range of empty weights is only 2 ton (~5%). Unlike the broad solution curve for TOGW, the OWE solution curve is relatively confined, like the wetted area solution curve.

Fig. 15. OWE vs Splan of convergence solutions.

Fig. 16 shows that the fuel weight ranges from 190 ton to 240 ton for the two minima. The fuel weight decreases with $\tau$: the curve is steeper for higher $\tau$ values: by reducing $\tau$ from 0.2 to 0.19 about 50 ton are saved, while from 0.11 to 0.12 only 10 ton are saved. The takeoff gross weight changes between the two minima (i.e., the minimum planform area and the minimum weight), by 50 ton: TOGW~310 ton for $\tau=0.14$ and ~360 ton for $\tau=0.16$.

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The ff ranges from 0.61 ($\tau=0.14$) to 0.69 ($\tau=0.16$). This shows that for this mission the vehicle is fuel-dominated.

The Isp in this preliminary (and preliminary only) analysis is 2000 sec averaged over the mission. This value should be achievable with the propulsion hardware that has been developed during the past 40 years.

The TO wing loading is very consistent with a practical runway takeoff, as shown in Fig. 17.
A TO wing loading \( \sim 350 \text{ kg/m}^2 \) (71.7 lb/ft\(^2\)) is well within a practical value for a medium slender lifting body design\(^{10}\).

The fuel volume (see Fig. 17) is very high with respect to the typical vehicle hydrocarbon configurations \( \sim 2500 \text{ m}^3 \) to \( \sim 3500 \text{ m}^3 \), i.e., 60% of the total volume. Even with the drag increase due to increased volume, hydrogen provides a hypersonic range factor equivalent to a large subsonic cruise transport.

6. Conclusions

This analysis has shown that, notwithstanding the really challenging mission requirements, it is possible to define a range of possible solutions for the European LAPCAT II project. In fact, for the same M, range and payload requirements, a solution space exist rather than one (unique) solution. The takeoff gross weight ranges between \( \sim 310 \text{ ton} \) for \( \tau=0.14 \) to \( \sim 360 \text{ ton} \) for \( \tau=0.16 \); the fuel weight ranges from 190 ton to 240 ton; the ff ranges from 0.61 (\( \tau=0.14 \)) to 0.69 (\( \tau=0.16 \)). This shows that for this mission the vehicle is fuel-dominated. The solution of choice results from constraints that depend on mission typology (e.g., commercial aircraft, military aircraft, launcher...). In fact, for a commercial aircraft, passenger safety and comfort (acceleration) constraints are key requirements, as well as airport constraints.

The future work will be to define a trajectory and to impose aircraft passengers constraints and calculate a realistic converged configuration from TO to landing.

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