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SIZING OF SCRAMJET VEHICLES

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The current European project LAPCAT II has the ambitious goal to define a conceptual vehicle capable of achieving the antipodal range Brussels–Sydney ($\sim$ 18,000 km) in about 2 h at Mach number Ma = 8. At this high speed, the requirement of high lift to drag (L/D) ratio is critical to high performance, because of high skin friction and wave drag: in fact, as the Mach number increases, the L/D ratio decreases. The design of the vehicle architecture (shape and propulsion system) is, as a consequence, crucial to achieve a reasonably high L/D. In this work, critical parameters for the preliminary sizing of a hypersonic airbreathing airliner have been identified. In particular, for a given Technology Readiness Level (TRL) and mission requirements, a solution space of possible vehicle architectures at cruise have been obtained. In this work, the Gross Weight at Take-Off (TOGW) was deliberately discarded as a constraint, based on previous studies by Czysz and Vanderkerkhove [1]. 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. In order to screen the solutions found, requirements for taking-off (TO) and landing as well as the trajectory have been accounted for. A consistent solution has finally been obtained by imposing typical airliner constraints: emergency take-off and landing. These constraints enable singling out a realistic design from the broad family of vehicles capable of performing the given mission. This vehicle has been obtained by integrating not only aerodynamics, trajectory, and airliner constraints, but also by integrating the propulsion system, the trimming devices and by doing some adjustments to the conceptual vehicle shape (i. e., spatular nose). Thus, the final vehicle is the result of many iterations in the design space, until performance, trajectory, propulsion systems, and airport constraints are successfully met.

1 INTRODUCTION

Studies on hypersonic configurations in USA, Russia, and EU date back to the early 1960s. The lesson learned in the past [1–5] is that hypersonic vehicle sizing

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is very different from that for subsonic and supersonic aircraft [6–11]. Previous studies by Czysz [3] have shown that integrating individually optimized system elements yields a significantly reduced performance. In supersonic aircraft, each component is independently sized, designed, and assembled; in particular, the design of the vehicle begins by drawing constant wing area or constant weight aircraft concepts. In hypersonic vehicles, instead, sizing begins from mission distance and payload, not by drawing constant wing area or constant TOGW aircraft. Significant differences between conventional and hypersonic aircraft are the huge propellant volume (it is true, especially, for liquid hydrogen (LH_2) fuel) and the low aerodynamic efficiency in terms of the L/D ratio.

The sizing approach followed here is based on the so-called VDK/HC [1] parametric sizing methodology. This methodology was developed since the 1980s and applied to: high-performance subsonic to hypersonic aircraft; and reusable space launchers. For a cruising vehicle, the sizing begins with the mission distance, payload, and cruise Mach number to obtain a figure of merit (the Kuechemann's τ) for the whole vehicle. The VDK sizing methodology is based on the simultaneous solution for the OWE (overall weight empty) and planform area S_{plan} equations, ensuring that the separately calculated **available** and **required** weights and total volumes (V_{tot}) converge for a given τ [2], defined as $\tau = \left(V_{\mathrm{tot}}/S_{\mathrm{plan}}^{1.5}\right)$.

Note that all sizing 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 raises that of all systems and of the structure. The same occurs for the propellant volume: increasing its volume raises drag, and to keep the L/D ratio reasonable, a larger planform surface is needed to produce higher lift. However, a larger planform area means a more wetted area hence the structural weigh increases too, and the larger TOGW 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 (Ma, 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.

For a given mission requirements, more than one configuration can be found, and it is the constraints of mission typology (commercial aircraft, space plane, launcher, etc.) that will eventually define the "best configuration."

2 VEHICLE DESIGN

In [5], the present authors have found a solution space of aircraft configurations for given design specifications (cruise Ma = 8, range = 18.728 km, number of passengers = 300 ($W_{\mathrm{pay}} = 60$ t), and hydrogen fuel) by solving simultaneously all "cruise" equations.

At cruise, the initial *guess* to define a tentative conceptual vehicle consisted of $I_{\mathrm{sp}} = 2000$ s and engine thrust-to-weight T/W = 8.3. The variables related to the current state of industrial technology (I_{str}) were also assumed [3]: $I_{\mathrm{str}} = 21$ and 22 kg/m^2, $W_{\mathrm{sys}}/\mathrm{TOGW} = r_{\mathrm{sys}} = 0.07$ (W_{sys} is the weight of all systems), $\eta_v = 0.7$ (useable volume ratio).

At convergence, the solutions were found for four reference configurations (blended, elliptical cone, half elliptical cone, and Nonweiler-type waverider).

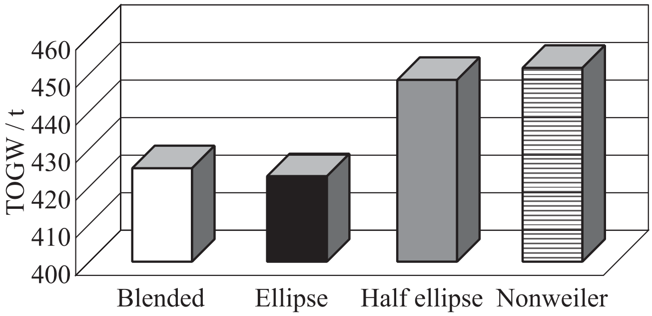


Figure 1 Weight esimates for different configurations

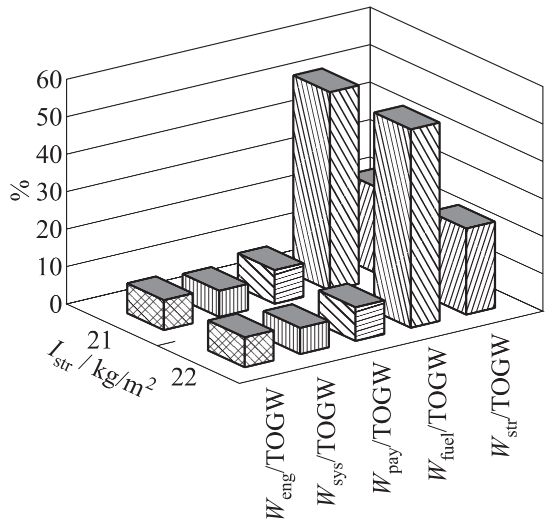


Figure 2 Weight distribution for $I_{\mathrm{str}} = 21$ and 22 kg/m^2

Comparing the weight estimates for the different configurations (Fig. 1), the ellipse and blended body have been found to be the most promising, that is, predicting the lowest TOGW ($\sim$ 420 t).

Figure 2 shows the weight distribution among the vehicle main components for a blended body configuration. This figure shows that for this (extreme) mission, the vehicle is fuel dominated. In Fig. 2, $W_{\rm eng}$ is the engine weight. Note, these results apply only to the cruise phase: the trajectory has not yet been included. The next step is then to define a trajectory and recalculate the converged configuration including take-off and landing.

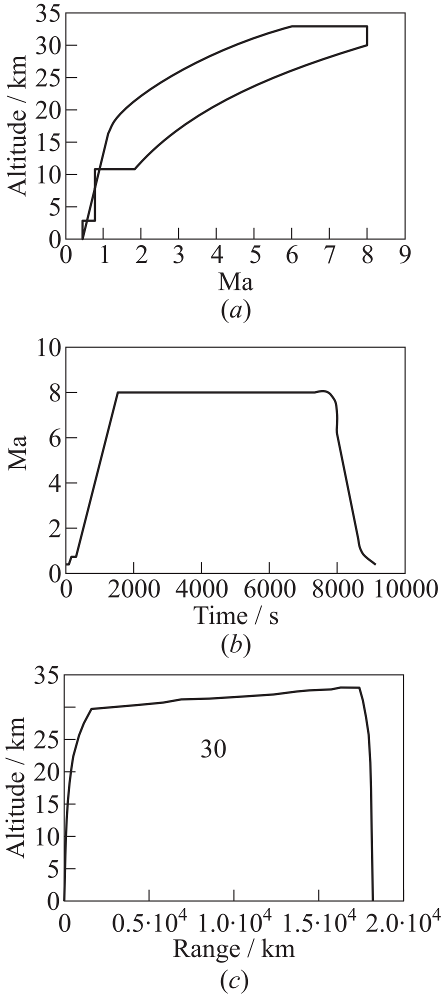
3 PRELIMINARY TRAJECTORY SELECTION

The reference trajectory has been calculated by means of a Numerical Energy Method. This method [3] involves the linearization of the equations of motion in order to obtain closed-form expressions for the desired performance parameters. These expressions are applied over finite velocity intervals where the aerodynamics, propulsion and flight path parameters are assumed to be constant. The method is extended by a rapidly converging iteration procedure to estimate climb performance for a flight path limited by sonic boom considerations and assuming the following (during *climb-out*):

- constant velocity climb-out to 3048 m;
- constant altitude acceleration to Mach 0.8;
- constant Mach 0.8 climb to 11,000 m;
- acceleration to the maximum dynamic pressure;
- constant dynamic pressure climb to 30,000 m;
- cruise, including climb to the maximum altitude; and
- maximum L/D descent.

Figures 3*a* to 3*c* show the reference trajectory. This trajectory has a profile similar to HyFAC and HyCAT studies [12, 13]. Limiting acceleration to $0.3g$, the time to climb is 14.4 min, cruise time is of 106.9 min, and descent time is 26.6 min.

Figures 4*a* to 4*c* show L/D, C_{D0} (drag coefficient), and $I_{\rm sp}$ (specific impulse) calculated along the trajectory. Lift-to-drag ratio shows a minimum between Ma = 0.8 and 2. In this range, the thrust delivered by the engines (ejector ramjets with an engine-thrust-to-weight (ETW) ratio equal to 22) must match that required by the vehicle. This is crucial to choosing the number of engines.



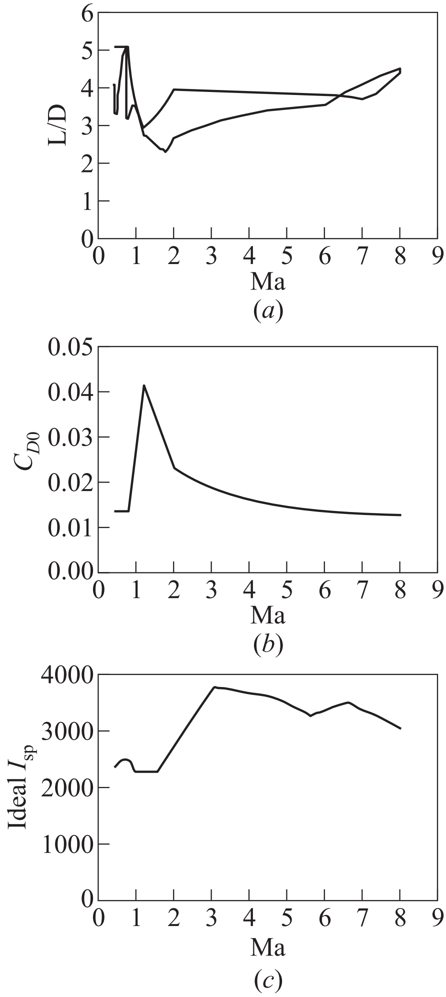


Figure 3 Reference trajectories: (a) altitude vs. Mach number; (b) Mach number vs. time; and (c) altitude vs. range

Figure 4 Lift-to-drag ratio (a); drag coefficient (b); and specific impulse (c) calculated along the trajectory

An ideal I_{sp} is calculated to begin estimating vehicle performance: this I_{sp} is a preliminary guess, as it does not account for control surfaces, engine drag, and the extra L/D due to the propulsion system.

With this I_{sp}, the time fraction to climb and descend is shown in Fig. 5, indicating that the faster the flight, the greater the fraction of time consumed for climb and descent. Reducing the speed by 25% reduces the climb and de-

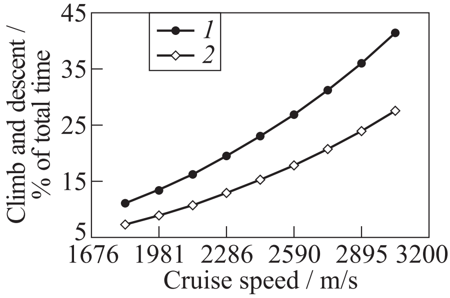


Figure 5 Climb and descent time fraction vs. cruise speed for 12,000- (*1*) and 18,000-kilometer (*2*) range

scent fuel consumed by 50% (though it increases the cruise consumption by 15%–20%)!

Once calculated, the total fuel fraction, ff, that is for climbing, cruise and descent, mass budget, and geometry must be, of course, recalculated.

4 SPACE SOLUTION OF HYPERSONIC AIRCRAFT

Once defined a preliminary trajectory, the preliminary vehicle configuration along this trajectory is also defined but fuel consumption, fuel and gross weights, actual $I_{\rm sp}$, and L/D ratio must be recalculated. The preliminary estimates above will be used as initial values to recalculate all variables for the entire realistic mission, from TO to landing. Once this is done, the trajectory is also recalculated, and this procedure is repeated until design specifications goals are met.

The weight has been iterated until converging with that calculated from the minimum volume requirement equations. Converged solutions for different structural indices were found in order to evaluate the solution trend as a function of $I_{\rm str}$. Sensitivity analysis (Figs. 6 and 7) shows that going from lower to advanced technology (higher to lower $I_{\rm str}$), all curves translate downwards. This is due to the fact that, as $I_{\rm str}$ decreases, the structural weight decreases and so does the TOGW. Then less $W_{\rm fuel}$ is needed, with positive effects on TOGW and total volume.

The minimum $S_{\rm plan}$ is 850 m^2 for $I_{\rm str} = 13$ kg/m^2 ($\tau = 0.18$) and 980 m^2 for $I_{\rm str} = 21$ kg/m^2 ($\tau = 0.18$): between these two $S_{\rm plan}$ minima almost 1000 m^3 of total volume are saved. Figure 6*b* shows that there are two minima, i. e., a minimum planform area and a minimum wetted surface: $\tau = 0.18$ corresponds to a solution with a minimum $S_{\rm plan}$, while $\tau = 0.15$ corresponds to a minimum $S_{\rm wet}$.

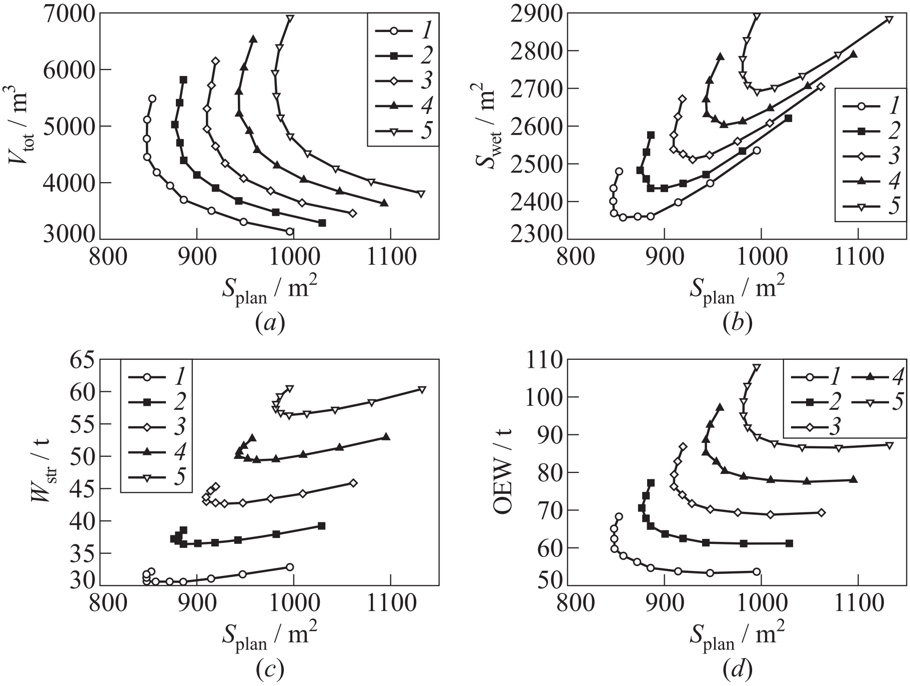


Figure 6 Total volume (V_{tot}) (a), S_{wet} (b), W_{str} (c), and OWE (d) vs. S_{plan}: *1* — I_{str} = 13 kg/m^2; *2* — 15; *3* — 17; *4* — 19; and *5* — I_{str} = 21 kg/m^2

These two solutions are very close; for example, for I_{str} = 21 kg/m^2, they go from about 2750 m^2 at 970 m^2 to 2700 m^2 at 1000 m^2, but there is still a range of solutions in-between to choose from.

A minimum S_{wet} means a minimum structural weight, i. e., 56 t at τ = 0.15. Figure 6c shows that the structural weight, W_{str}, has two closely spaced minima that are very reasonable at high skin temperature cruise (when the mass of the TPS is significant): in fact, at the minimum S_{plan} (corresponding to τ = 0.18), the structural weight is only 2 t more than that of the minimum S_{wet}.

Figure 6c also shows that going from I_{str} = 21 to 13 kg/m^2, the structural weight decreases by about 26 t, going from 56 to 30 t for the two S_{wet} minima; almost 700 m^3 are saved between these two S_{wet} minima.

Going from I_{str} = 15 to 21 kg/m^2, the Operational Weight Empty (OWE) varies by about 40 t for τ = 0.15 (corresponding to the minimum S_{wet}). The TOGW goes from 275 to 350 t, saving 75 t. Figure 7b shows that for I_{str} = 21 kg/m^2, the OWE range is between 205 to 207 t for the two minima: the range of empty weights is only 2 t ($\sim$ 1%). Unlike the broad solution locus

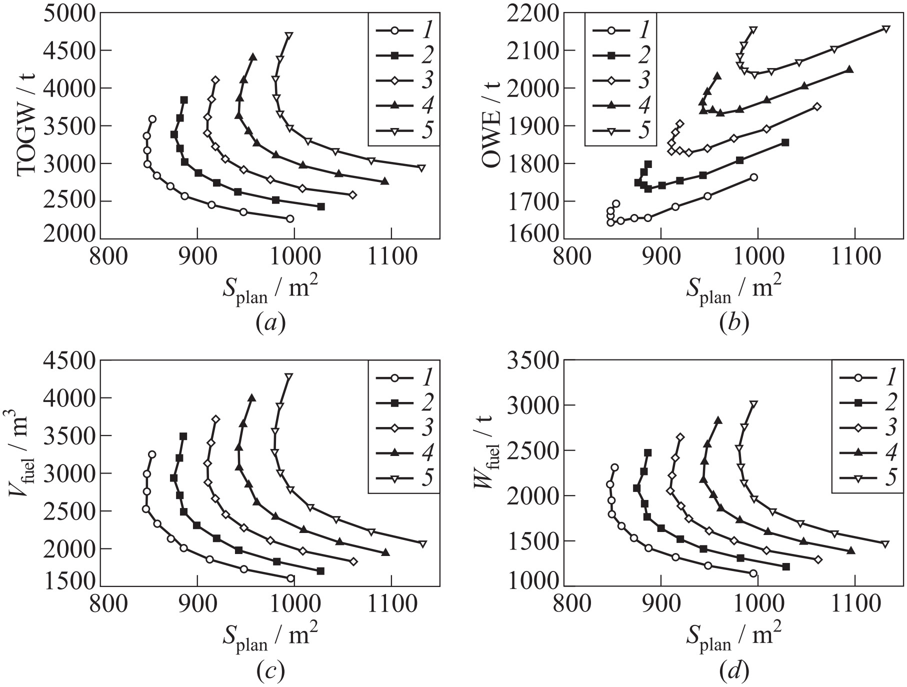


Figure 7 Cross weight at take-off (TOGW) (a), OWE (b), $V_{\rm fuel}$ (c), and $W_{\rm fuel}$ (d) vs. $S_{\rm plan}$: *1* — $I_{\rm str} = 13$ kg/m^2; *2* — 15; *3* — 17; *4* — 19; and *5* — $I_{\rm str} = 21$ kg/m^2

curve for TOGW, the OWE solution curve is relatively short, like for the wetted area.

Figure 7d shows that going from $I_{\rm str} = 21$ to 15 kg/m^2, the $W_{\rm fuel}$ saves about 50 t: from 200 to 150 t between the two $S_{\rm wet}$ minima and from 180 to 230 t between the two $S_{\rm plan}$ minima.

The fuel weight ranges from 150 to 300 t for $I_{\rm str} = 21$ kg/m^2. The fuel weight decreases with τ: the curve is steeper for higher τ: by reducing τ from 0.2 to 0.18, about 50 t are saved, while from 0.1 to 0.13, only 10 t are saved.

The TOGW changes between the two minima (i. e., the minimum planform area and the minimum weight), by 50 t: TOGW ~ 310 t for $\tau = 0.15$ and ~ 360 t for $\tau = 0.18$.

The ff ranges from 0.61 ($\tau = 0.15$) to 0.69 ($\tau = 0.18$). This shows that for this mission, the vehicle is fuel-dominated.

The TO wing loading is very consistent with a practical runway take-off as shown in [3]. A TO wing loading ~ 350 kg/m^2 is well within a practical value for a medium slender lifting body design.

5 COMMERCIAL AIRCRAFT CONSTRAINTS AND VEHICLE SELECTION

Once found a hypersonic vehicle space solution, commercial aircraft constraints [14] must be accounted for the selection of the best solution within the range of convergence. In particular:

- for passenger comfort: limit axial acceleration $a \leq 0.3g$; and
- compliance with important JAR field performance requirements means:
  - TO with one engine inoperative climb requirement [14];
  - emergency landing with high fuel load ($\mathrm{CL}_{\max}$, W/S where W = TOGW and S is S_{plan}); and
  - runway length = 3048 m (as for a B-747).

The final space solution (in terms of TOGW vs. S_{plan}) obtained from this iterative process is shown in Fig. 8. It is seen that τ goes from 0.10 to 0.24. The minimum TOGW for $I_{\mathrm{str}} = 21$ is 550 t for $\tau = 0.12$ and $S_{\mathrm{plan}} = 1550$ m^2. A minimum S_{plan} is 1300 m^2 for $\tau = 0.18$. Imposing a maximum landing weight (MLW) of 70% TOGW for the emergency landing condition, the solution is found below $\tau = 0.16$. Because decreasing τ below 0.12 does not lower either W_{fuel} or TOGW, it is worthless to consider solutions below $\tau = 0.1$. The appropriate range of solutions lies, thus, between $\tau = 0.12$ and 0.14.

Table 1 shows the weights and geometry of the vehicle calculated from TO to landing. A conceptual shape for this schematic but realistic vehicle is shown in Fig. 9.

Actually, the solution just found is not the final configuration because it is a simple elliptical cone-shape configuration: control surfaces and engine vehicle-integration must and will be sized in a follow-on future paper.

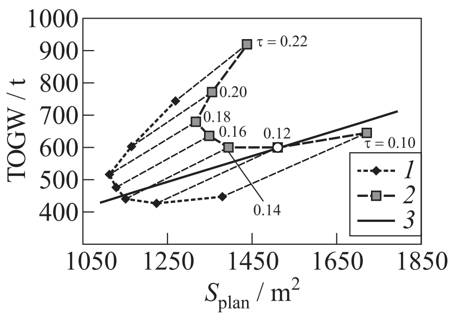


Figure 8 Cross weight at take-off vs. S_{plan}: *1* — $I_{\mathrm{str}} = 17$ kg/m^2; *2* — $I_{\mathrm{str}} = 21$ kg/m^2; and *3* — $W/S = 395$ kg/m^2, MLW

Table 1 Converged solutions with $I_{str} = 17$ and 21 kg/m^2

I_{str}, kg/m^2	17	21
Geometry		
τ	0.16	0.12
S_{plan}, m^2	1134.63	1503.07
b, m	31.70	36.49
c, m	6.34	7.30
Λ, m	59.65	68.66
h, m	4.20	3.63
Weight		
TOGW, kg	476,529	599,091
OWE, kg	150,189	228,386
W_{pay} kg	29,256	29,256
OEW, kg	119,461	197,658
W_{fuel}, kg	326,340	370,705
W_{str}, kg	56,721	99,860
ff	0.68	0.62
W_{str}/TOGW	0.119	0.167
Volume		
V_{total}, m^3	150,189	228,386
V_{pay}, m^3	510.0	510.0
V_{fuel}, m^3	4372.8	4967.2
V_{ENG}, m^3	329.1	511.5
V_{sys}, m^3	321.3	338.9
V_{fix}, m^3	199.0	199.0

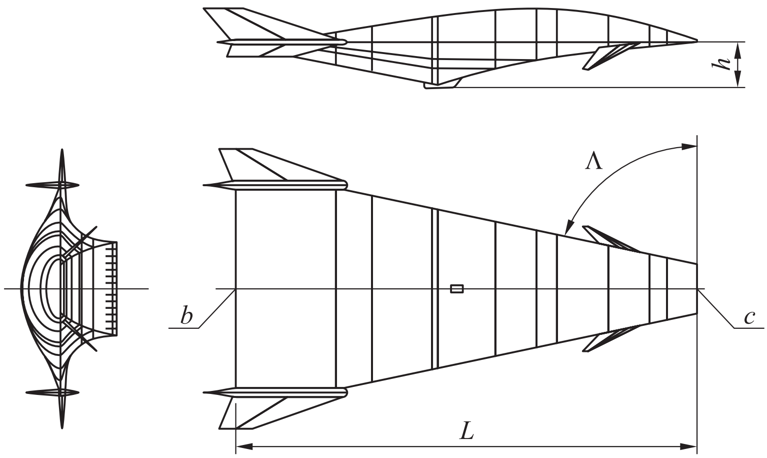


Figure 9 Vehicles configuration

6 CONCLUDING REMARKS

This analysis has shown that, notwithstanding the really challenging mission requirements, it is possible to define a range of possible solutions for the LAPCAT II vehicle. Further, it has shown that a conservative structural technology may be selected without a dramatic impact on vehicle size. In fact, fuel weight and volume requirements in conjunction with the emergency TO and landing wing area requirements, are the primary drivers of aircraft size. Structural and payload weight are of secondary importance in comparison.

Given the large impact of fuel weight and volume on the total vehicle size, care must be taken to ensure that the I_{sp} and thrust goals are met for the engine propulsion system.

At first analysis, the $I_{\mathrm{str}} = 17$ kg/m^2 aircraft is selected as the baseline vehicle design due to its moderate structural technology level and the minor weight savings with respect to a more technologically mature $I_{\mathrm{str}} = 15.1$ kg/m^2 vehicle.

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