Sizing of Scramjet Vehicles

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Abstract

The current 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 ratio 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.

In this work, for given technology levels (TRL) and mission requirements critical parameters for a preliminary sizing of an hypersonic airbreathing airliner were identified. In particular, a solution space of possible vehicle architecture capable of meeting mission "cruise" conditions were obtained. In order to screen these solutions, also requirements for taking-off (TO) and landing as well as the trajectory have been accounted for. Once a space solution of conceptual vehicles is obtained, a individual solutions can be obtained by imposing typical airliner constraints. 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 has been obtained by not only integrating aerodynamics, trajectory and airliner constraints, but also by integrating the propulsion system, the trimming devices and by doing some adjustments to the vehicle conceptual shape (i.e. spatular nose). In fact, this airliner is the result of many iterations in the design space, until performance, trajectory, propulsion systems and constraints are successfully achieved and met.

In this work, the Gross Weight at Take Off (GWTO) was deliberately discarded as a constraint, based on previous studies by Czysz et al. [1, 2, 5]. Typically, limiting from the beginning the GWTO 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.

1. Introduction

Studies on hypersonic configurations in USA, Russia and EU date back to early sixties. The lesson learned in the past is that hypersonic vehicle sizing is very different from that for subsonic and supersonic aircraft [6-11]. Previous studies by Czysz [5], have shown that the approach of integrating individually optimized system elements across matching yielded a significant reduction in performance. 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. In hypersonic vehicles instead 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 sizing approach followed here is based on the VDK/HC [1] parametric sizing methodology. This methodology was developed since the '80s and applied to: (a) high-performance subsonic to hypersonic aircraft; and, (b) reusable space launchers. Sizing begins with the mission distance, payload and cruise Mach (=Ma) number, to obtain a figure of merit (the Kuechemann's tau) for the whole vehicle. The VDK sizing methodology is based on the simultaneously solution for the *OWE* (overall empty weight) and planform area S_{pln} equations, ensuring that the separately calculated available and required weights and volumes converge for a given tau [3], defined as

$$\left(\tau = \frac{V_{tot}}{S_{plan}^{1.5}}\right).$$

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 L/D reasonable a larger planform surface is needed to produce higher lift. But a larger planform area means more wetted area, the structural weigh increases too, and the larger take-off gross weight (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 was found, and it is the constraints of mission typology (commercial aircraft, space plane, launcher…) that will define the "best configuration".

2. Vehicle Design

In work [4] by these authors, a solution space of aircraft configurations for given design specifications: cruise Ma=8, range=18,728 km, number of passengers = 300 (Wpay=60 ton), and hydrogen fuel, was found by solving simultaneously all 'cruise' equations.

At cruise, the performance assumed to define a first tentative conceptual vehicle were: Isp=2000 s, engine T/W=8.3; the variables related to the state of current industrial technology have also been fixed [5] to: Istr=15, 18, and 21 kg/m^2, $W_{sys}/W= r_{sys}=0.07$ (W_{sys}= weight of all systems), $\eta_v=0.7$ (useable Volume).

Converged solutions were found for five reference configurations (wing body, blended, elliptical cone, half elliptical cone and waverider). Comparing the weight estimates for the different configurations results (see Fig. 1 and Fig. 2), the blended body has been found to be the most promising, that is, with the lowest TOGW (415 ton) and W_{fuel}.

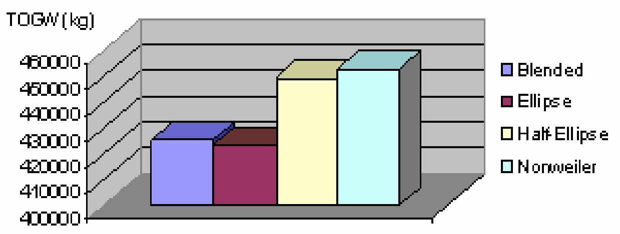

Fig. 1 TOGW for the different configurations

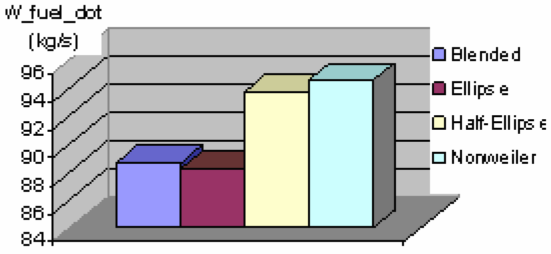

Fig. 2 W_{fuel} for the different configurations

Fig. 3 and Fig. 4 show the weight and the weight percentage distribution among the vehicle main components. These figure show that for this [extreme] mission, the vehicle is fuel dominated.

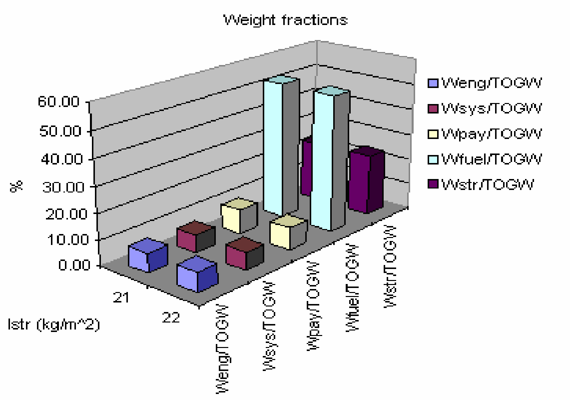

Fig. 3 Weight distribution

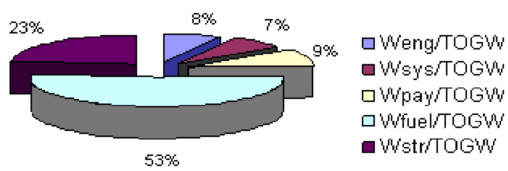

Fig. 4 Weight percentage (Istr=21, 22 kg/m2)

These results refer only to cruise: the trajectory has not been included. The next step is then to define a trajectory and calculate again a converged configuration from TO to landing.

3. Commercial aircraft constraints and trajectory selection

The reference trajectory has been calculated by means of a Numerical Energy Method. The method [5] involves the linearization of the equations of motion in order to obtain closed-form expressions for the desired performances 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 convergent iteration procedure to estimate climb performance for a flight path limited by sonic boom considerations and assuming:

- during *climb-out: 1.* constant velocity climb-out to 3048 m, 2. constant altitude acceleration to Mach 0.8, 3. constant Mach 0.8 climb to 11,000 m, 4. acceleration to max dynamic pressure; *constant dynamic pressure climb to 30,000 m; cruise including climb to maximum altitude; maximum L/D descent*.

Fig. 5 - Fig. 7 show the reference trajectory . This trajectory has a profile similar to HyFAC and HyCAT studies. With an acceleration of 0.3 g, time to climb is 14.4 min, cruise time is of 106.9 min and descent time 26.6 min.

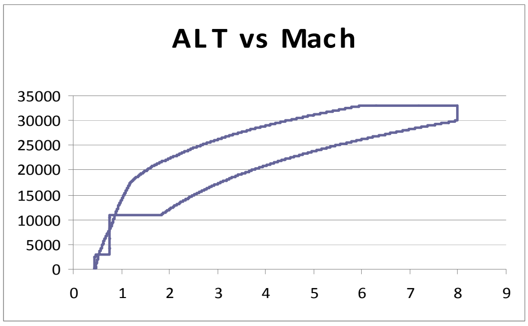

Fig. 5 Altitude vs Mach

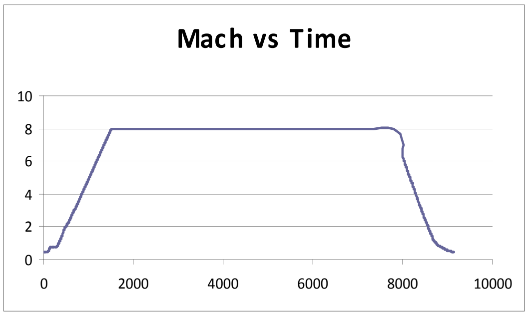

Fig. 6 Mach vs time

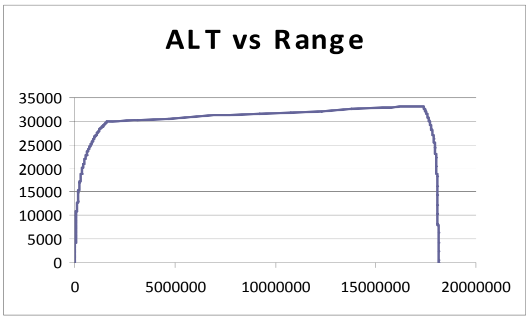

Fig. 7 Altitude vs Range

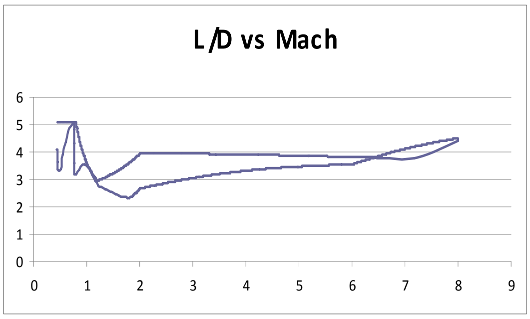

Fig. 8 L/D vs Mach

Fig. 8 - Fig. 10 show vehicle performance calculated along its trajectory.

L/D shows a minimum between Ma 0.8 and 2. In this range, the thrust furnished by the engines (turboramjets) must supply that required by the vehicle. This is a crucial point in choosing the number of engines.

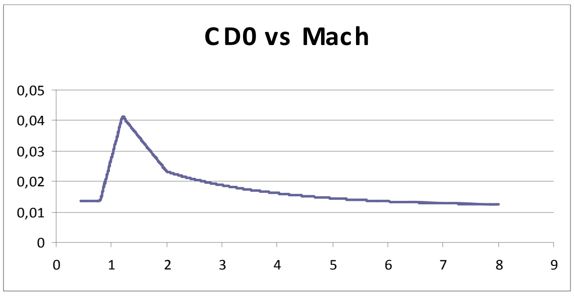

Fig. 9 CD0 vs Mach

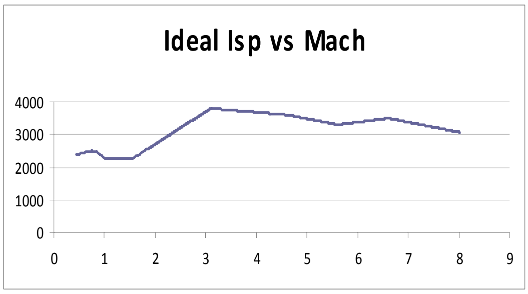

Fig. 10 Ideal Isp vs Mach

An ideal Isp is calculated for the first conceptual vehicle estimate: this Isp is preliminary, and expedient, as it does not account for control surfaces, engine drag and the extra L/D due to the propulsion system.

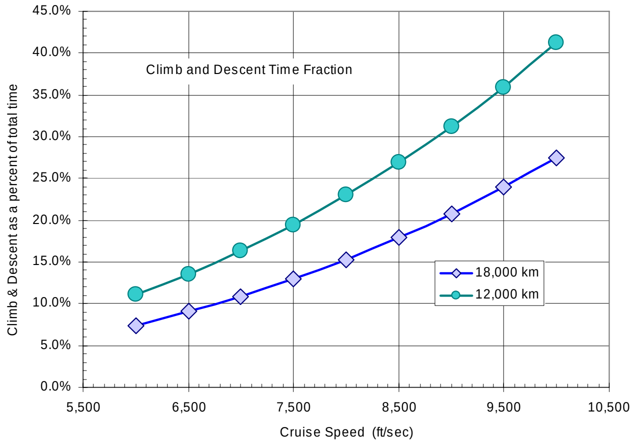

Fig. 11 Climbing and landing time vs cruise speed

For this trajectory, the time fraction of climb and descent is shown in Fig. 11, showing that the faster the flight, the greater the fraction of time consumed in climb and descent. Reducing the speed by 25% reduces the climb and descent fuel consumed by 50%!

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

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* Isp and L/D must be re-calculated. They will be implemented as starting values to calculate all variables for the entire mission, from TO to landing. This done, the trajectory is again recalculated and the 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 indeces were found in order to evaluate the solution trend as a function of Istr. Sensitivity analysis (see Figs. 12-19) shows that going from lower to advanced technology (higher to lower Istr) all curves translate downwards. This is due to the fact that as Istr decreases the structural weight decreases and so does the TOGW . Thus lower Wfuel is needed, with positive effects on TOGW and the total volume.

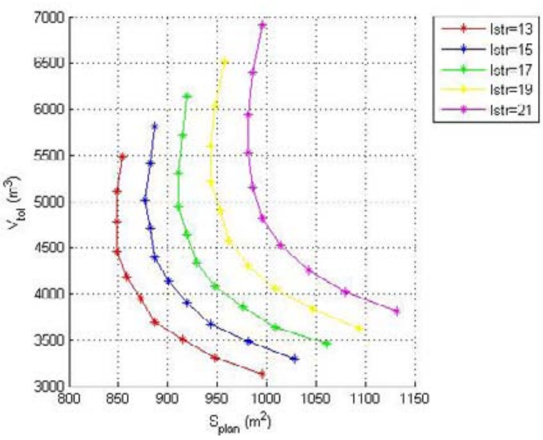

Fig. 12 V_{tot} vs Splan

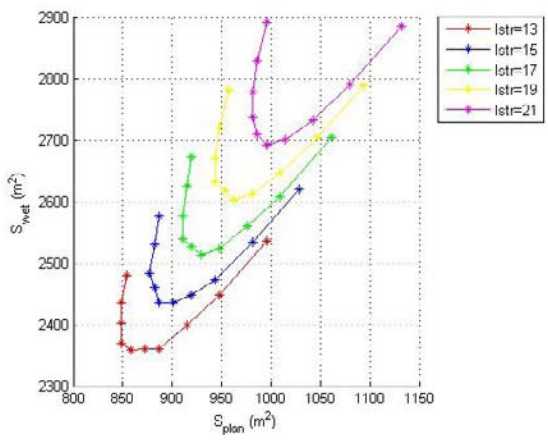

Fig. 13 S_{wet} vs Splan

The mimimum Splan is 850 m^2 for Istr=13 (tau=0.16) and 980 m^2 for Istr=21 (tau=0.16). Between the two Splan minima almost 1000 m^3 are saved. Fig. 13 shows there are two minima, a minimum planform area and a minimum wetted surface. tau=0.16 corresponds to a solution with a minimum Splan, while tau=0.14 correspond to a minimum Swet. These solutions are very close, for example for Istr=21, they go 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_{wet} means a minimum structural weight, i.e. 56 ton at tau=0.14 and 2 ton extra at tau=0.16. The structural weight, Wstr, has two closely spaced minima that are very reasonable at high skin temperature cruise (when the mass of the TPS is significant).

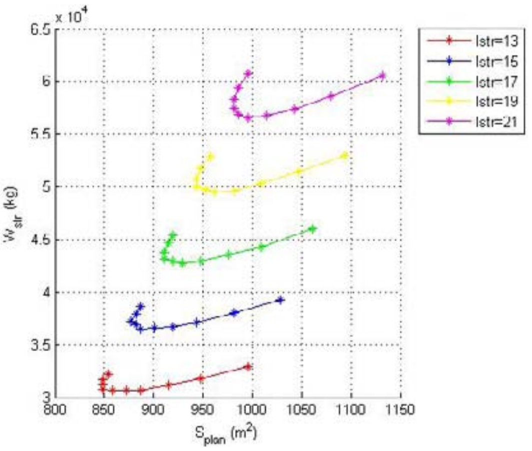

Fig. 14 W_{str} vs Splan

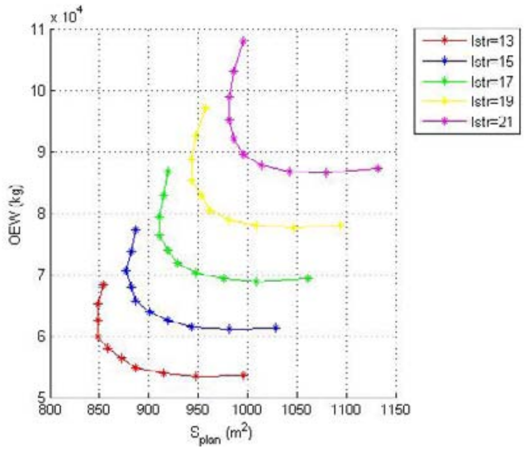

Fig. 15 OEW vs Splan

Going from Istr=21 to Istr=13, the structural weight decreases by about 26 ton, going from 56 ton to 30 ton for the two structural weight minima. Almost 700 m^3 are saved between these two Wstr minima.

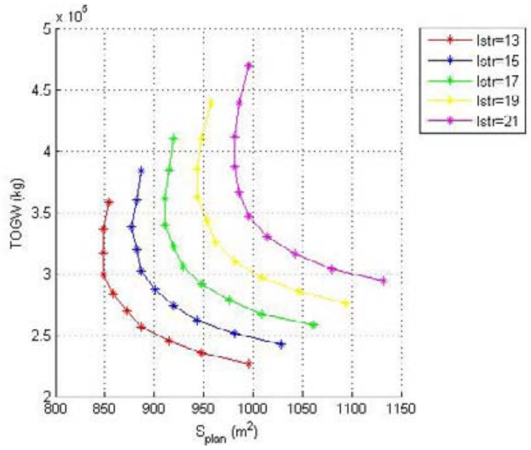

Fig. 16 V_{tot} vs Splan

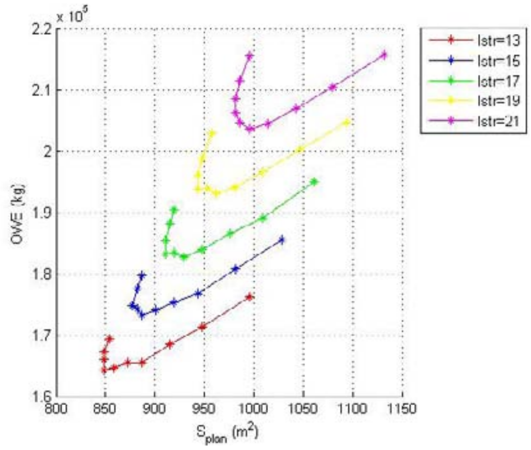

Fig. 17 V_{tot} vs Splan

For a given Istr, the Operational Empty Weight (OWE) varies by about 40 ton between these two minima. The TOGW goes from 275 to 350 ton, saving 75 ton. This OWE shows a minimum 205 ton for tau=0.15. A typical OWE range is between 205 ton to 207 ton for the two minima: the range of empty weights is only 2 ton (~1%). Unlike the broad solution curve for TOGW, the OWE solution curve is relatively confined, like the wetted area solution curve.

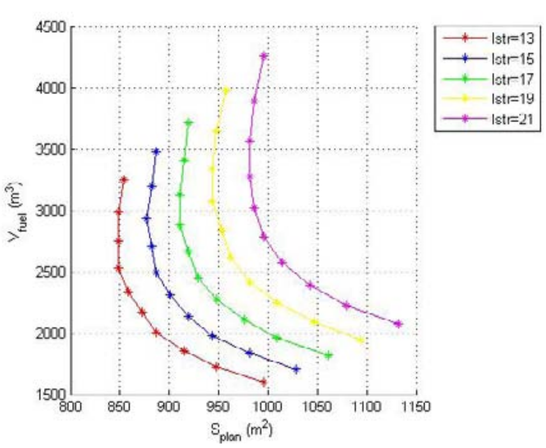

Fig. 18 V_{fuel} vs Splan

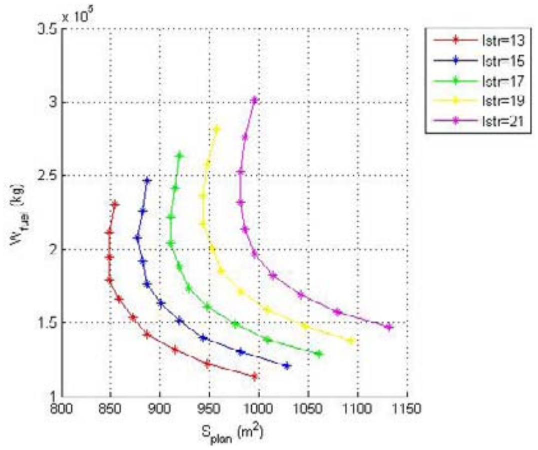

Fig. 19 W_{fuel} vs Splan

The reduction in Wfuel saves about 50 ton (from 200 to 150 ton) between the two Wstr minima and 180 ton to 230 ton between the two Splan minima.

The fuel weight ranges from 150 ton to 300 ton for Istr=21. The fuel weight decreases with tau: the curve is steeper for higher tau: by reducing tau from 0.20 to 0.18 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.

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 TO wing loading is very consistent with a practical runway takeoff, as shown in [5].

A TO wing loading ~350 kg/m^2 (71.7 lb/ft^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 [12] must be accounted for the selection of the best solution within the range of convergence. In particular:

1. for passenger comfort: horizontal acceleration a≤ 0.3 g
2. compliance with important JAR field performance requirements, that means:
   a. take-off (TO) with one engine inoperative (OEI) climb requirement [12]

b. emergency landing with high fuel load (CLmax, W/S,)

c. runway length = 10,000 ft (as for a B 747,)

The final space solution given by the iterative process is shown in Fig. 20.

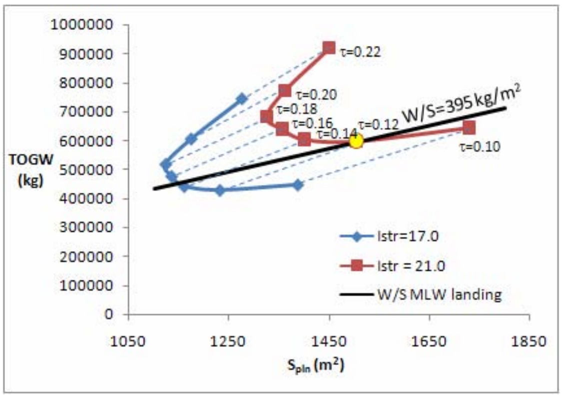

Fig. 20 TOGW vs Splan

Fig. 20 shows that tau goes from 0.10 to 0.24. The minimum TOGW for Istr=21 is 550 ton for tau=0.12 and Splan=1550 m^2. A minimum Splan 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 tau below 0.12 neither Wfuel nor TOGW decrease, it is not useful to consider solutions below tau=0.1. The appropriate range of solutions lies then between tau=0.12 and 0.14.*

Tab. 1 shows weights and geometry of the vehicle calculated from TO to landing.

A conceptual shape, for this schematic but realistic vehicle is shown in Fig. 21. 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.

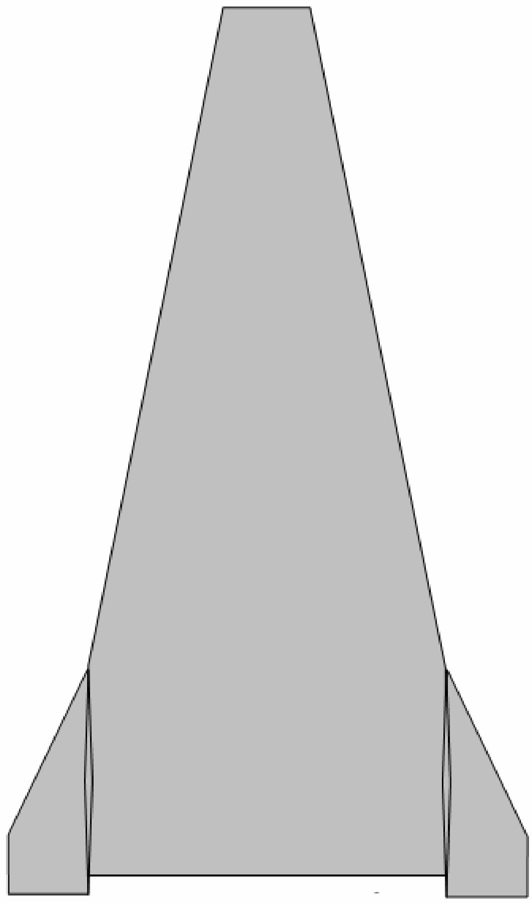

Fig. 21 Vehicles configuration with Istr=18

Istr (kg/m^2)	17	21
Geometry		
t	0.16	0.12
Spln (m^2)	1134.63	1503.07
b (m)	31.70	36.49
c (m)	6.34	7.30
L (m)	59.65	68.66
h (m)	4.20	3.63
Weight		
TOGW (kg)	476529	599091
OWE (kg)	150189	228386
Wpay (kg)	29256	29256
OEW (kg)	119461	197658
Wfuel (kg)	326340	370705
Wstr (kg)	56721	99860
ff	0.68	0.62
Wstr/TOGW	0.119	0.167
Volume		
V_{total} (m^3)	150189	228386
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

Tab. 1 Converged solutions

6. Conclusions

This analysis has shown that, notwithstanding really challenging mission requirements, it is possible to define a range of possible solutions for the European LAPCAT II vehicle. Further, this analysis has shown that a conservative structural technology level 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 air TBCC+SCRJ propulsion system.

At first analysis, the I_{str} = 18.0 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_{str} = 15.1 vehicle.*

Acknowledgements

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