CONCEPTUAL DESIGN METHODOLOGY FOR LOW-SUPERSONIC LH2-POWERED PASSENGERS AIRCRAFT

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Abstract

Commercial aviation domain has always been characterized by a constant technological evolution, featuring different progress slopes depending on the historical period, but still focused on the development of more efficient and competitive products. More recently, the concept of competitiveness has changed, moving from a pure performance standpoint to a wider concept, encompassing economic viability and environmental sustainability. The latter is crucial and cannot be neglected anymore, even for a sector that is contributing to manmade CO2 emissions only by 2-4%. The need to assess the overall product lifecycle in terms of environment compatibility is thus influencing new design for conventional aircraft, with particular focus on Sustainable Aviation Fuels (SAF) and hydrogen. However, this shall not limit technological progress, which is basically stuck around jet airliners configuration concepts originally developed around 1950s. As a matter of fact, still maintaining the need of environmental compatibility, supersonic and, in general, high-speed aviation has regained momentum in terms of research efforts and funding, after some period characterized by highs and lows since the withdrawn from service of the Concorde. This paper aims at evaluating the possibility of designing a liquid hydrogen-powered aircraft aimed at passengers transportation, flying in low supersonic regime along trans-Atlantic routes, with particular focus on the feasibility of Concorde-like configurations modified to host liquid hydrogen on-board. Particularly, a conceptual design methodology is described and results are discussed to highlight limitations of such kind of aircraft layout, when non drop-in fuels are exploited.

Keywords: Supersonic aircraft, High-speed aviation, Liquid hydrogen, Sustainable aviation, Conceptual design

1. Introduction

Commercial aviation domain has always been characterized by a constant technological evolution, featuring different progress slopes depending on the historical period, but still focused on the development of more efficient and competitive products. More recently, the concept of competitiveness has changed, moving from a pure performance standpoint to a wider concept encompassing economic viability and environmental sustainability. Aviation decarbonization is in fact a mandatory target in order to reach the goals identified with the 2015 COP 21 Paris agreement [1] and adopted, for example, by ICAO through the Carbon Offsetting and Reduction Scheme for International Aviation (CORSIA) [2] and further extended through the Long Term Aspirational Goal (LTAG) [3] published by its Committee on Aviation Environmental Protection (CAEP). For what concerns conventional aviation, different strategies can be put in place, depending on the type of aviation segment, on the readiness level of the infrastructure and on the technologies. As described in [4], commuter and very short range aircraft routes may benefit from the introduction of all-electric or hybrid-electric powerplant, while short to medium range platforms appear to be one of the best candidates for the exploitation of hydrogen, as fuel either to power hydrogen-hybrid powerplant or to feed pure hydrogen turbine-based engines. For the same segment, the adoption of Sustainable Aviation Fuel (SAF) can be also considered, while it will be the most probable solution for long range and very long range transport, considering the amount of fuel required and the limited volume dedicated to fuel storage on-board. On the other hand, while maintaining the need of environmental compatibility, supersonic and, in general, high-speed aviation has regained momentum in terms of research efforts and funding, after some period characterized by highs and lows since the withdrawn from service of the Concorde. New supersonic aircraft concepts are in fact under design [5], while international airworthiness agencies such as FAA and EASA are starting to publish initial guidelines for the establishment of a stable regulation for supersonic aircraft, aimed at setting up certification criteria [6]. As far as European landscape is concerned, sustainable aviation researches are being funded within the Horizon 2020 funding scheme, with relevant activities associated to, but not limited to, projects such as "(LTO) noiSe and EmissioNs of supErsoniC Aircraft" (SENECA) [7] and "MDO and REgulations for Low boom and Environmentally Sustainable Supersonic aviation" (MORE&LESS) [8, 9] aimed at assessing noise, pollutant/greenhouse gas emissions of supersonic aircraft during flight and on Landing & Take-Off cycles (LTO), as well as effects on climate.

Considering the aforementioned topics, it is interesting to assess whether hydrogen can be effectively used not only to power conventional subsonic platforms, but also to feed supersonic transports by means of direct combustion of the propellant, substituting conventional Jet A1 fuel or SAF. In fact, exploiting hydrogen would completely reduce CO2 emissions in operation to zero, since hydrogen combustion does not produce CO and CO2, even if water vapor, NOx and contrails generation shall be still assessed to evaluate potential benefits. On the other hand, hydrogen would allow the reduction of life cycle-related CO2 emissions only if produced by green sources (otherwise CO2 emissions would be simply moved to an early stage of its life, before operation), which in turn represent, at the moment, the most expensive methods of production, influencing its cost per unit kg [10-11]. On the design perspective, storage of hydrogen, even in its liquid form (LH2) is a challenge, both because of the storage temperature (20K, cryogenic) but also because of its low energy density per unit volume that would require considerable amount of space on-board (even though the overall mass would be reduced because of the high energy per unit mass). This is even more critical for Concorde-like configurations, such as Boom Overture [5], where the space is limited and wing volume cannot be used because of its very limited size, which is not suitable to host rigid tanks that shall be fitted with insulation as well (on the contrary, traditional drop-in fuels may use integral tanks). For this reason, this paper primarily aims at evaluating the possibility of designing a LH2-powered aircraft, aimed at passengers transportation, flying in the range of Mach 1.5 – 2.5 in cruise along routes that can be compatible with those flown by Concorde (6000 km), carrying 60 to 100 passengers. Particularly, a Mach 2 concept is studied first, with reference to Concorde operational scenario. Afterwards, a sensitivity analysis on Mach number and payload is performed to evaluate the main differences, advantages and disadvantages related to the exploitation of hydrogen, with focus on the impact on configuration. After this brief introduction, Section 2 lists the high-level requirements driving the design problem, as well as potential case studies to be considered. Section 3 presents the conceptual design methodology, with focus on aerodynamic prediction by analytical methods, mass breakdown estimation, as well as performance evaluation. As innovative step within the conceptual design process, the evaluation of volume required to store the LH2 is introduced as integral part of the sizing iteration, in order to neglect unfeasible configurations because of volume inconsistencies. Section 4 presents the main outcomes and results of the analysis, while Section 5 draws major conclusions.

2. Problem statement and case studies

In order to start the design process, a first set of high-level requirements is defined so to identify a potential reference design point to be reached. However, as already anticipated in Section 1, the methodology aims at assessing a family of aircraft concepts within the low-supersonic regime (cruise Mach 1.5 – 2.5) starting from the reference vehicle, so also a range of variation for some basic mission-related performance is specified. Particularly, a sensitivity analysis on Mach number in cruise and on number of passengers will be performed in order to assess the impact on aircraft configuration. The typical range is not modified, since a larger one would require an excessive increase of dimensions of the aircraft, while a lower one would not be in line with a potential market of operation, considering that, in this case, the vehicle would not be able of flying trans-Atlantic routes. With current limitations inhibiting flight over inhabited lands, because of sonic boom problems, this would inevitably make the concept economically unfeasible. The situation is clarified in Tables 1 and 2. All aircraft will exploit LH2 as propellant and a Concorde-like, delta-wing configuration.

Table 1 – Reference high-level requirements

Data	Value
Passengers capacity	80
Mach number in cruise	2
Typical range [km]	6000

Table 2 – Range of applicability of sensitivity analysis with reference to initial requirements

Data	Min. Value	Max. Value
Passengers capacity	60	100
Mach number in cruise	1.5	2.5

3. Conceptual design methodology

3.1 Overall workflow

The overall design process is based on the workflow described in this section. Considering that some similarities with conventional aircraft concepts still remains, even with the peculiarities of high-speed regimes and LH2 storage, a typical iterative approach based on [12] is selected as main rationale. In fact, the baseline method is applicable to conventional subsonic aircraft, but it remains accurate also for low-supersonic vehicles if specific elements are introduced to characterized these kinds of configurations. Also, the iterative approach is still built on the identification of a consistent take-off mass, derived from initial evaluations about aerodynamics of the configuration as well as from mass fractions computation along the mission profile, and obtained after design process convergence. The iteration converges more easily when take-off mass is lower, so the rational appears favorable for LH2-powered aircraft, which are intrinsically characterized by a reduced take-off mass with reference to kerosene-based competitors. So, as shown in Figure 1, after an initialization of relevant variables, which could be done by means of a statistical database of reference vehicles or simply by a mathematical definition of consistent first-guess values for the iterating variables, the mission profile is hypothesized. Then, preliminary estimations start with the aerodynamic analysis of the configuration, followed by mass breakdown evaluation, performance requirements verification during the mission (in terms of thrust-to-weight ratio and wing loading) and concept validation after assessing the feasibility of LH2 storage. The update of take-off mass depending on the newly derived mass and volume breakdowns, influenced by aerodynamic performance and lifting surfaces (constraining the layout of the vehicle) allows comparing the derived weight with the initial (guess) value, and, if this is within the specified tolerance margin, the iteration reaches convergence. Otherwise, the process continues. The fact that the LH2 storage volume verification is directly embedded within the cycle allows ensuring the feasibility of the concept. The results will thus be already capable of hosting the hydrogen, being potentially characterized by a larger planform dimensions with reference to kerosene competitors, especially for what concerns fuselage size. In fact, typically, volume verification is not really a critical step within conceptual design processes for conventional aircraft, considering that, with current fuel consumption targets and wing shapes, the volume on-board is typically enough to satisfy mission requirements.

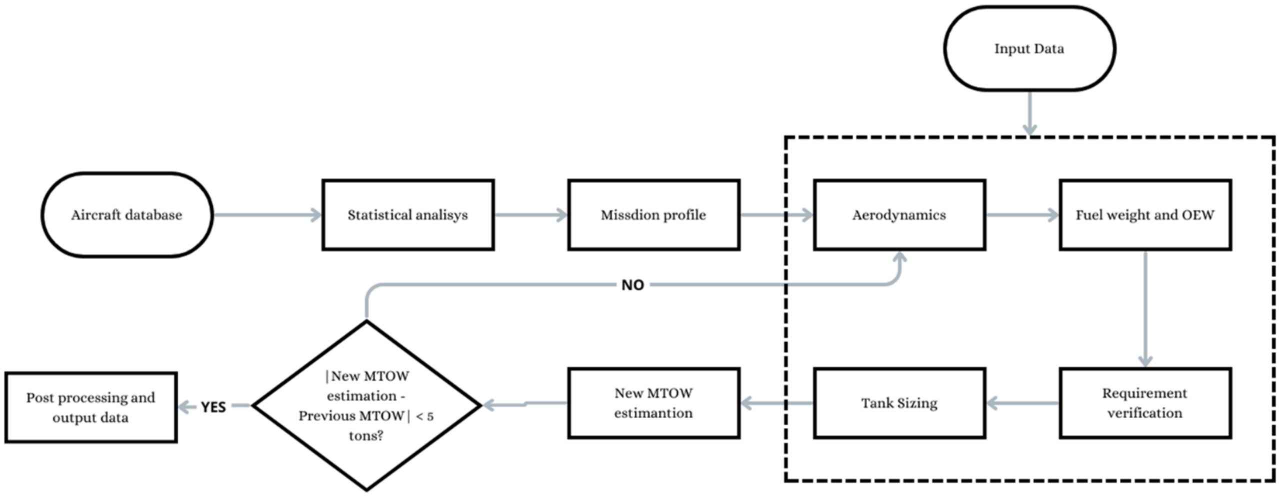

Figure 1 – Conceptual design workflow

As reference mission, the case study flying at Mach 2 is following the profile shown in Figure *2*. The other aircraft combinations will have similar mission concepts, properly modified to match high-level requirements specified in Section 2.

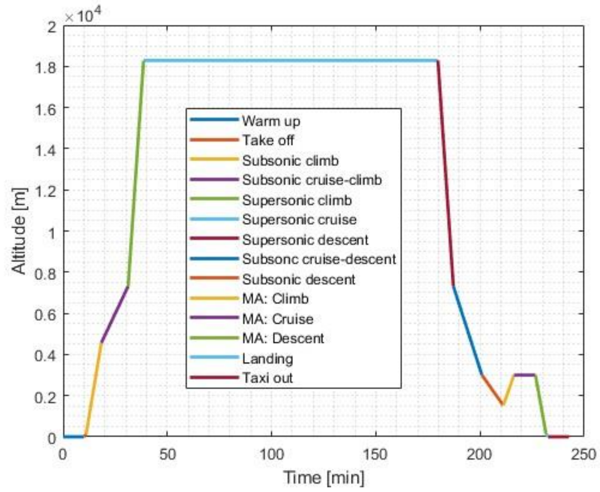

Figure 2 – Reference mission profile (Mach 2 case study)

The profile features a first subsonic climb, after ground and take-off phases, followed by an additional cruise/climb trajectory to reach inhabited areas. Then the supersonic climb starts and continues up the cruise level. In a symmetrical way, the descent profile follows the same layout, with a final approach to the destination which is interrupted once, in order to take into account for a Missed Approach phase (MA). Then, the final (successful) landing attempt is performed, concluding the mission after a short ground roll. A Mach equal to 0.80 is hypothesized for subsonic cruise/climb segments, while a progressive acceleration from take-off up to 0.80 is selected for low altitude climb. Similarly, the Mach number ranges from 0.80 to cruise Mach number for final supersonic climb. The descent profile is, again, symmetrical.

Following subsections describe the different algorithms used within the conceptual design process, while Section 4 reports the numerical results of the different runs, both for reference aircraft and derived platforms.

3.2 Aerodynamic analysis

In this preliminary evaluation stage, aerodynamic analysis is performed using analytical and semi-empirical algorithms for the estimation of lift and drag coefficients directly at vehicle level. According to [12], lift coefficient C_L can be estimated through the product of lift curve slope coefficient C_{L_α} and the angle of attack α, where slope parameter is defined as in (1) for subsonic and in (2) for supersonic conditions (the latter being very simplified).

$$C_{L_{\alpha_{SUB}}} = \frac{2\pi \cdot AR}{2+\sqrt{4+\left(\frac{AR\cdot\beta}{0.95}\right)^2\cdot\left(1+\left(\frac{\tan\Lambda}{\beta}\right)^2\right)}} \cdot \frac{S_{exposed}}{S_{ref}} \cdot F \qquad (1)$$

$$C_{L_{\alpha_{SUP}}} = \frac{4}{\sqrt{M^2-1}} \qquad (2)$$

Where AR is the aspect ratio, Λ is wing sweep in radians, $\beta = \sqrt{1-M^2}$ and M is the Mach number.

For what concerns subsonic regime, fuselage lift factor F is defined as in (3)

$$F = 1.07 \cdot \left(1 + \frac{d_{ext}}{b}\right)^2 \qquad (3)$$

With d_{ext} as the external fuselage diameter and b as the wingspan in meters.

Surfaces ratio in this case has been defined considering that the exposed outer surface of the delta wing facing the airflow is approximately a triangle, while the section covered by the fuselage is treated as a rectangular shape. The calculated exposed surface pertains to half of the wing (either top or bottom), thus to obtain the total exposed planform area, it needs to be doubled. The segment covered by the fuselage is subsequently added to S_{ref} (total planform area), while not considered for $S_{exposed}$.

Drag coefficient C_D is a combination of parasite and induced drag, with general formulations, always suggested by [12], reported in (4) and (5) for both subsonic and supersonic regimes.

$$C_D = \frac{\sum C_{f_i} \cdot FF_i \cdot Q_i \cdot S_{wet_i}}{S_{ref}} + C_{D_{misc}} + C_{D_{L\&P}} + C_{D_{i_{SUB}}} \qquad (4)$$

$$C_D = \frac{\sum C_{f_i} \cdot S_{wet_i}}{S_{ref}} + C_{D_{wave}} + C_{D_{misc}} + C_{D_{L\&P}} + C_{D_{i_{SUP}}} \qquad (5)$$

Friction coefficient C_f is calculated as in (6), where a turbulent flow is hypothesized, according to the definition of Reynolds and Cut-Off Reynolds defined in (7) and (8) for subsonic and supersonic correlations respectively,

$$C_{f_{turb}} = \frac{0.455}{(\log(\min(Re, Re_{cutoff})))\text{^}2.58 \cdot (1+0.144 \cdot M^2)\text{^}0.65} \qquad (6)$$

$$Re = \frac{\rho \cdot V \cdot l}{\mu} \qquad (7)$$

$$Re_{cutoff} = 38.21 \cdot \left(\frac{l}{k}\right)^{1.053} \qquad (8)$$

where ρ is the density in kg/m^3, V is the speed in m/s, μ is the dynamic viscosity in $Pa \cdot s$, l is the characteristic length and k is the skin roughness value in m.
For subsonic regime, form factors FF_i and interference drag factors Q_i (which need to be computed, in parallel with related S_{wet_i} for wing, tail and other main aircraft elements, such as intakes) are computed according to the suggestions provided in [12], together with the values of miscellaneous

as well as leakages and protuberances drag $C_{D_{misc}}$ and $C_{D_{L\&P}}$.

Drag due to lift C_{D_i} is computed using the leading edge suction method, expressed as in (9) and (10) for both subsonic and supersonic conditions.

$$C_{D_{i_{SUB}}} = \left(\left(\frac{S}{\pi \cdot AR} \right) + \left(\frac{1+S}{C_{L_{\alpha_{SUB}}}} \right) \right) \cdot C_{L_{SUB}}^{2} \qquad (9)$$

$$C_{D_{i_{SUP}}} = \left(\left(\frac{S}{C_{L_{\alpha_{SUP}}}} \right) + \left(\frac{1+S}{C_{L_{\alpha_{SUP}}}} \right) \right) \cdot C_{L_{SUP}}^{2} \qquad (10)$$

Where S is the percentage of wing leading edge suction as function of design C_L as reported in [12]. Ultimately, wave drag C_{D_W} is computed from the dimensional relationship expressed in (11) and dividing by the planform area S_{ref}.

$$\left(\frac{D}{q\infty} \right)_{wave} = 1.2 \cdot \left(1 - 0.2 \cdot (M - 1.2)^{0.57} \cdot \left(1 - \frac{\pi \cdot \Lambda^{0.77}}{100} \right) \right) \cdot \left(\frac{9}{2} \pi \cdot \left(\frac{A_{Max}}{l} \right)^{2} \right) \qquad (11)$$

where q_∞ is the dynamic pressure in Pa, A_{Max} is the estimated maximum cross-section area of the vehicle in m^2.

All values associated to aircraft dimensions and surfaces are subjected to update within the iteration cycle, thus the aerodynamic estimations are also updated along the process, being consistent with the growing size of the vehicle, up to convergence.

3.3 Mass breakdown derivation

Mass breakdown derivation for the vehicle is based on the identification of proper empty mass and fuel mass according to the approach provided by [12]. Payload and crew masses are instead known. Empty mass M_{Empty} in kg can be statistically computed according to equations such as the one shown in (12), as function of iterating take-off mass M_{TO}.

$$\frac{M_{Empty}}{M_{TO}} = 0.97 \cdot M_{TO}^{-0.06} \qquad (12)$$

Fuel mass is instead function of mission phases. For cruise, it is possible to exploit the Breguet formulation (13), once aerodynamic efficiency L/D and specific fuel consumption SFC are known for a specific cruise range R and speed V.

$$\frac{M_{cruise_{end}}}{M_{cruise_{start}}} = e^{-\frac{R \cdot SFC \cdot g}{V \cdot \frac{L}{D}}} \qquad (13)$$

For phases where altitude is not constant, it is possible to use mass fractions to hypothesize the amount of fuel burnt, as reported in (14-17). According to [10], cruise fraction already includes descent contribution, so this is not specifically defined here. Also, it is worth noting that fuel fractions provided in [12] are typically used for kerosene-based aircraft. This means that the values are higher if compared to what can happen to LH2-powered aircraft, because of the lower fuel mass with reference to vehicle mass of the latter. The estimation is thus updated, and the fractions reported in (14-17) are assumed.

$$\frac{M_{taxi_{end}}}{M_{taxi_{start}}} = 0.999 \quad (14)$$

$$\frac{M_{sub-climb_{end}}}{M_{sub-climn_{start}}} = 0.995 \quad (15)$$

$$\frac{M_{sup-climb_{end}}}{M_{sup-climb_{start}}} = 0.995 \quad (16)$$

$$\frac{M_{landing_{end}}}{M_{landing_{start}}} = 0.999 \quad (17)$$

The final fuel mass M_{fuel} in kg can be computed according to (18), where reserves are also included. The combination of the overestimation of fuel consumption due to the reserve factor, allows to take into account the missed approach defined within the mission.

$$M_{fuel} = 1.06 \cdot \left(1 - \frac{M_{mission_{end}}}{M_{mission_{start}}}\right) \cdot M_{TO} \quad (18)$$

Take-off mass is thus the combination of empty mass, fuel mass, as well as payload and crew mass, being a function of aircraft characteristics, such as aerodynamic efficiency and size, so it is updated along the cycle.

3.4 Preliminary performance evaluation

In order to conclude the preliminary vehicle evaluation, a verification of the concept with reference to performance-related requirements for the main mission phases is necessary. This is implemented through the matching chart analysis [13-15], which is in this case tailored for supersonic aircraft. The matching analysis allows defining a proper design point for the vehicle in terms of thrust-to-weight ratio (T/W) as function of wing loading (W/S), within a 2D graph named matching chart. Equations and curves reported in the chart are representative of equilibrium conditions during the flight, conceived to verify aero-propulsive balance of the concept in different regimes. For the purpose of this study, requirements for take-off, second segment, climb (subsonic/supersonic), cruise (subsonic/supersonic), landing and turn are expressed, so to derive a proper design point for the vehicle under study. Two charts are considered to separate subsonic and supersonic conditions (even if the powerplant still remains the same for both conditions).

Notably, take-off run requirement is represented by equation (19)

$$\left(\frac{T}{W}\right)_{TO} = \frac{W/S}{TOP \cdot \sigma \cdot C_{L_{TO}}} \quad (19)$$

where

$\frac{W}{S}$ is the wing loading in kg/m^2

TOP is the take-off parameter as defined in [10]

σ is air density ration, depending on airport elevation

$C_{L_{TO}}$ is lift coefficient at take-off

Second segment phase can be represented by the requirement shown in equation (20), as defined within the regulation [16] in terms of minimum climb gradient G_{2S} to be guaranteed in case of failure to one of $N_{engines}$.

$$\left(\frac{T}{W}\right)_{2S} = \left(\frac{N_{engines}}{N_{engines}-1}\left(\frac{1}{\left(\frac{L}{D}\right)_{2S}} + G_{2S}\right)\right)1/\sigma \qquad (20)$$

Climb phase is modeled using the equilibrium equation (21), where an average condition can be defined for either subsonic or supersonic regime in order to maintain a desired climb gradient G_{climb} with a specific throttle Π (it is possible to select an average altitude/speed combination or top-of-climb/beginning-of-climb respectively, depending on the most critical condition). Still, at least two curves, one for subsonic and one for supersonic regimes shall be selected.

$$\left(\frac{T}{W}\right)_{climb} = \left(\frac{q_\infty C_D}{W/S \cdot g} + G_{climb}\right)\frac{1}{\Pi\sigma} \qquad (21)$$

Similarly to climb, also cruise condition can be modeled using equation (22), neglecting climb gradient. Also in this case, two equations shall be considered to take into account both subsonic and supersonic regimes (a general formulation is reported in (22)).

$$\left(\frac{T}{W}\right)_{cruise} = \left(\frac{q_\infty C_D}{W/S \cdot g}\right)\frac{1}{\Pi\sigma} \qquad (22)$$

Landing requirement (or stall requirement, depending on the reference condition) can be expressed as in (23) where a requirement on wing loading is reported as function of desired landing distance s_{LND}, multiplied by a safety factor (less the space required to clear the obstacle s_a), lift coefficient, aircraft mass ratio and airport elevation. The aircraft will satisfy this requirement if reference wing loading is lower than the derived value (i.e. if the wing surface is equal or larger, for a specific mass).

$$\left(\frac{W}{S}\right)_{landing} = \frac{1.67 \cdot s_{LND} - s_a}{5} \cdot C_{L_{LND}} \cdot \frac{M_{TO}}{M_{LND}} \cdot \sigma \qquad (23)$$

Ultimately, turn requirement can be expressed as in (24). Even though maneuverability is not a primary concern for this kind of aircraft, it is still possible to enrich the design space with an additional wing loading requirement, usually less critical than landing requirement, to take into account turns with contingency factor n.

$$\left(\frac{W}{S}\right)_{turn} = \frac{\frac{1}{2}\rho V^2 C_{L_{turn}} \sigma}{ng} \qquad (24)$$

Feasible conditions shall be identified within the area characterized by lower wing loading, with reference to smallest wing loading requirements, as well as by higher thrust-to-weight ratios, with reference to the highest curve.

Typically, most critical wing loading for this kind of vehicles is identified within landing or stall conditions, where the speed is low and a large wing area is required to sustain the vehicle (also considering the low aerodynamic performance in subsonic regime). This means that the wing surface is sized according to subsonic critical conditions. In supersonic conditions instead, this value becomes a constraint, since the wing cannot change its surface. A reference wing loading is thus selected, according to previously derived wing surface, to identify a design point for thrust-to-weight requirements in supersonic climb and cruise.

All the aspects here described will be discussed in more detail in Section 4.

3.5 Hydrogen storage

Usually, previous steps described in the above subsections are enough to sketch the layout of the reference aircraft, considering that mutual geometrical relationships can be derived depending on the family of vehicles selected. In this case, considering the need of storing LH2 within the fuselage, a further verification of the concept is needed, and it shall be based on the evaluation of fuel quantity to be stored within the available volume. This shall be an integral part of the iteration cycle, considering that it may constrain the configuration to have a larger or longer fuselage, depending on the criteria selected. Moreover, since integral tanks cannot be used and conformal tanks can be difficult to produce, rigid cylindrical or ellipsoidal tanks shall be considered for the storage, introducing volume reductions due to structural integrity and insulation of the compartments [17-18].

For this particular work, two LH2 compartments have been considered: one behind the passengers cabin, using most of the volume available within the fuselage, and one under passengers deck. The sizing procedure is thus constrained to this configuration, but it can be easily updated depending on user needs. Particularly, a preliminary estimation of structural thickness of the tank, as function of internal pressure and material, is provided, together with the analysis of required insulation to keep the fluid at thermal equilibrium at 20K. The overall tank volume will be thus estimated according to required fluid volume plus dry volume, defining volumetric and gravimetric efficiencies that will influence the conceptual design iteration process.

The underfloor tank is sized first. This has a fixed configuration featuring a parallelepiped shape with circular tank ends along length and width. The height of the tank is determined using a simple geometrical relation (25), where the internal diameter of the fuselage $d_{fus_{int}}$, cabin height h_{cabin}, cargo hold height h_{cargo} and a margin are considered to select the maximum suitable tank height h_{tank}. Similarly, the width of the tank is determined as well, depending on its vertical location inside the fuselage.

$$h_{tank} = d_{fus_{int}} - \left(h_{cabin} + h_{cargo} + h_{empty}\right) \qquad (25)$$

The length of the tank is constrained by the length of the cabin compartment.

The remaining fuel is instead stored within the rear cylindrical tank, using most of the internal fuselage volume.

Both tanks are sized in terms of structural and insulation layers using (26) and (27).

Particularly, equation (26) is derived from [19], allowing the iterative definition of wall thickness t_w as function of material properties (Maximum stress σ_f, Young modulus E_γ) and burst pressure p_p, considering a safety margin SM and cross section dimensions of the tank a, c (semi-axes, if this is elliptical).

$$\frac{\sigma_f}{SM} \geq p_p\left[\frac{a+c}{2t_w}\cdot\left(1+2\cdot\left(1+3.6\cdot\frac{p_p}{E_\gamma}\cdot\left(\frac{a+c}{2t_w}\right)^3\right)\cdot\frac{a-c}{a+c}\right)+0.5\right] \qquad (26)$$

Burst pressure is derived from design pressure, which in turn is function of pressure differential across the shell Δp (27).

$$p_p = 2\cdot p_{ultimate} = 2\cdot\left(1.5\cdot(1.1\cdot\Delta p)\right) \qquad (27)$$

Equation (28) is instead derived from [20], allowing for a quick estimation of passive insulation layer (and related thickness t_{ins}) required to maintain the fluid at reference 20K temperature.

$$t_{ins} = 2\cdot\sqrt{\frac{k_{ins}\cdot t_{fl}\cdot(T_{int}-T_{LH2})}{h_{g,LH2}\cdot\rho_{ins}}} \qquad (28)$$

Where

$k_{ins}$is the thermal conductivity of the material in W/mK

ρ_{ins} is insulation material density in kg/m^3

T_{int}, T_{LH2} are compartment temperature in which the tank is contained and LH2 temperature

t_{fl} is target flight time for which passive insulation is required in s

$h_{g,LH2}$ is heat of evaporation of LH2

The overall convergence loop is concluded when the concept is validated with reference to both performance and LH2 storage requirements, provided that the take-off mass has reached a stable value as well.

4. Results

This section lists the results of the iteration loops for both the reference aircraft and the related concepts derived through the sensitivity analysis. Main output are shown and discussed, together with matching charts and layout overview. Limitations of LH2 applications are highlighted, as well as feasibility problems, where identified.

4.1.1 Reference aircraft

The results of the analysis applied to the reference aircraft are here reported. As indicated in Table *1*, the aircraft is conceived to carry 80 passengers over a 6000 km route, with a Mach equal to 2 in cruise. The main assumptions reported in Table 3 are also considered.

Table 3 – Main input used for the analysis of reference aircraft

Variable	**Value**
Wing sweep angle [°]	60
Aspect ratio	1.8
Runway length at take-off and landing [m]	4000
Specific fuel consumption at Mach 2 [kg/h/daN]	0.30
Reference lift coefficient at take-off	1.5
Storage pressure for LH2 [MPa]	0.2
Max external temperature to be managed by tank [K]	323

Reference geometry factors for wing positioning and tail are assumed to be similar to Concorde architecture, as well as mission phases parametrization (climb gradients, flight levels, Mach numbers etc…). Hydrogen tanks are supposed to be made of a Aluminum 7075 alloy shell (480 MPa yield), featuring also Multi-Layered Insulation and vacuum chamber insulation assembly (0.3 mW/mK conductivity) around the case.

The main results for the converged solution are shown in Table 4 . Vehicle layout is shown in Figure 3, while details on cross-section views for the tanks are shown in Figure 4. Reference matching charts for subsonic and supersonic regimes are shown instead in

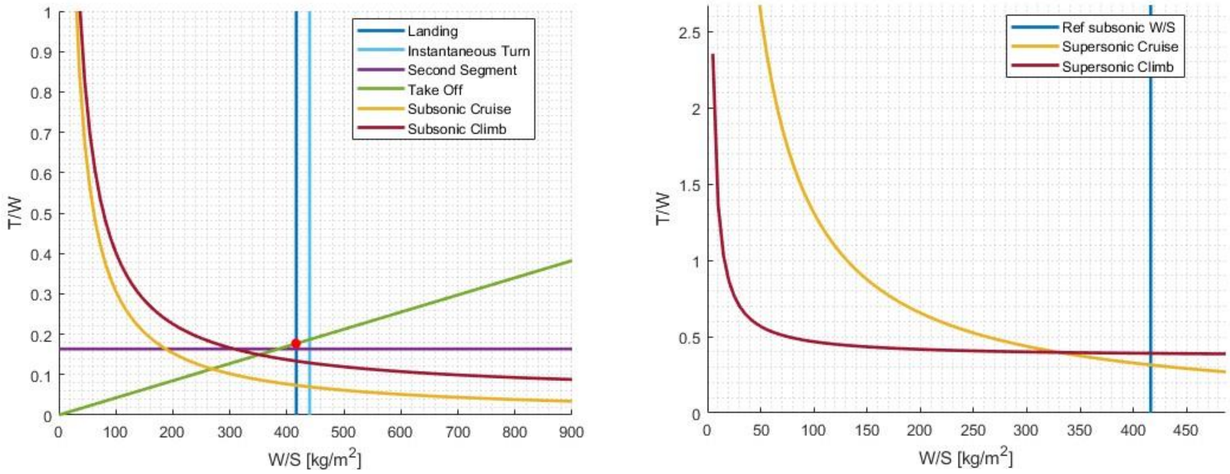

Figure *5*.

Table 4 - Main output for the reference aircraft

Variable	**Value**
Operating Empty Mass [kg]	101500
Fuel Mass [kg]	28900
Gross Take-Off Mass [kg]	139600
Wing surface [m2]	335
Wingspan [m]	28
Vehicle length [m]	71
Fuselage external diameter [m]	5
Fuel volume (total) [m3]	410
Rear tank volume [m3]	335
Underfloor tank volume [m3]	76
Wing loading (at gross mass) [kg/m2]	415
Required thrust-to-weight ratio in subsonic regime	0.20
Required thrust-to-weight ratio in supersonic regime	0.40
Aerodynamic efficiency at Mach 2	5

If compared to Concorde, the aircraft is almost 10 meters longer, with a higher empty mass. This is due to the extra airframe portion required to host the tanks and to the tanks themselves which add a considerable amount of mass (around 10000 kg). On the other hand, since the overall take-off mass is lower with respect to Concorde, the wing surface is also smaller, resulting in a higher wing loading. Required fuel mass is one third of the one used by Concorde (as it is reasonable to expect because of the properties of hydrogen which has three times the energy per unit mass of kerosene), but it requires more than 400 m3 for the storage. For this reason, the fuselage diameter (Figure 4) is constrained at 5 m maximum (in order both to limit the length of the vehicle, while maintaining a reasonable fineness ratio for the fuselage). An initial estimation of gravity center is provided, both at full quantity of fuel and in empty condition, considering the contribution of overall structure and payload. Tanks layout shown in Figure 4 are then taken as reference both for the Mach 2 case study and for the derived configurations for the same reason.

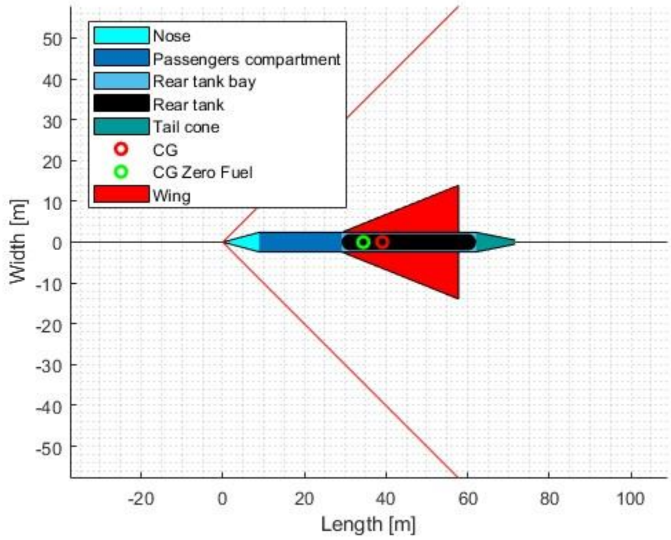

Figure 3 - Vehicle layout, plan view (reference aircraft)

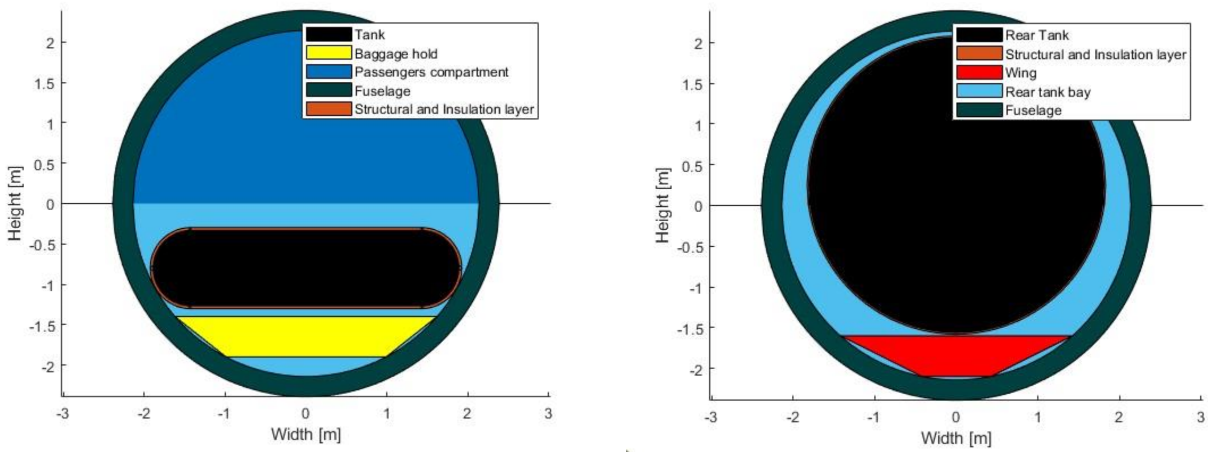

Figure 4 - Cross section view of LH2 tanks (left - underfloor tank, right – rear tank)

In terms of matching (Figure 5), the required thrust at take-off is much lower than the one available for the Concorde (around 140 kN dry for each engine), being limited in this case to 280 kN (total). However, for the purpose of this study, a longer take-off run is hypothesized (Table 3). Together with this, the reduction of take-off mass contributes lowering the thrust requirements in this phase. Supersonic thrust requirements are instead more demanding, considering the increase of cross-section of the aircraft (and the associated increase of drag). In fact, when translated at sea-level (as reported in Figure 5, right chart), the overall required thrust is around 135 kN per engine, suggesting that the supersonic climb (and related acceleration) is the most demanding phase for the powerplant. This is a typical result coming from the reduced slenderness of the aircraft, contributing to increase its thrust-to-drag ratio.
Overall, the aircraft concept presents the main advantage of being capable of flying the same route with a reduced mass at take-off, but, as disadvantage, a higher volume is required, with impact on fuselage dimensions. The dimensions are actually the cause for which the aerodynamic efficiency is limited to 5, with reference to the value of around 7 in supersonic regime for the Concorde.

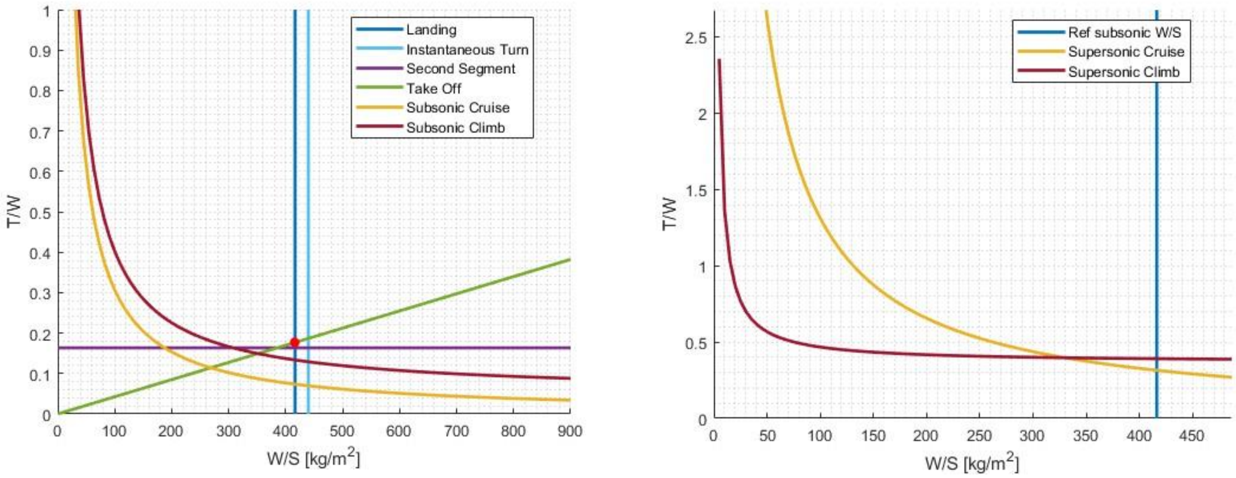

Figure 5 - Matching charts for the reference aircraft (left - subsonic regime, right – supersonic regime)

4.1.2 Sensitivity analysis

Considering the range of sensitivity shown in Table *2*, two case studies are here considered to validate the method and to provide potential alternative to the reference aircraft studied in Section 4.1.1. The first one consists in an aircraft able to carry 100 passengers on a 6000 km flight, with a Mach number in cruise limited to Mach 1.5. The second concept is instead conceived to carry 60 passengers on the same route, flying at Mach 2.5 in cruise. With the same input provided in Table *3* and an external diameter of the fuselage constrained to 5 m, it is interesting to evaluate the impact of Mach number in cruise with reference to the updated payload. Specific fuel consumption in cruise is modified, with a value of 0.28 kg/h/daN for the first case study, while 0.32 kg/h/daN is used for the second one. In order to reduce the negative impact of fuel storage needs on overall volume, the higher number of passengers is associated to the lower Mach number in cruise, producing the aforementioned aircraft alternatives. Reference mission profiles are updated as in Figure 6.

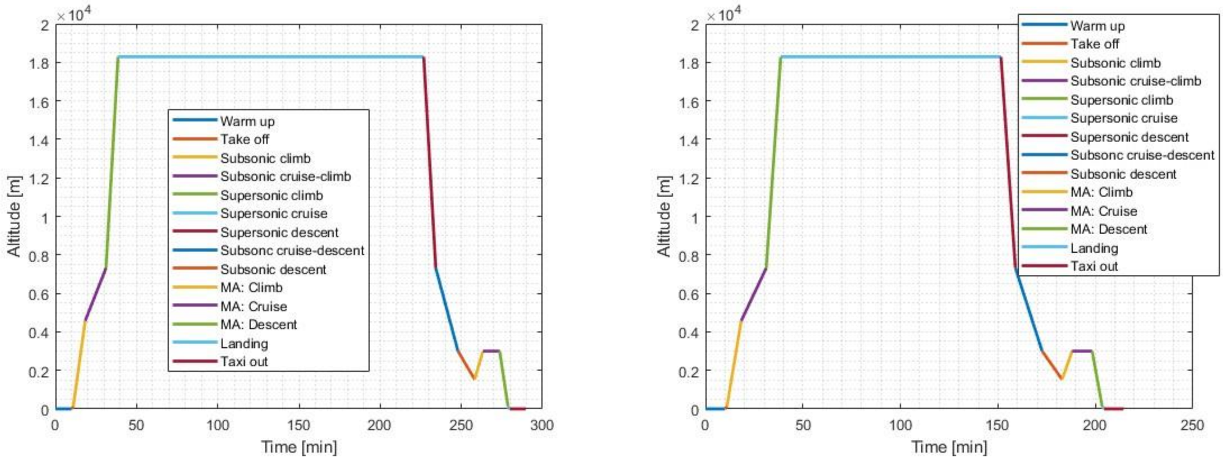

Figure 6 - Mission profiles for Mach 1.5 (left) and Mach 2.5 (right) case studies

The results for the first concept are shown in Table 5.

Table 5 - Main output for Mach 1.5 case study

Variable	Value
Operating Empty Mass [kg]	100000
Fuel Mass [kg]	26000
Gross Take-Off Mass [kg]	145000
Wing surface [m2]	340
Wingspan [m]	27
Vehicle length [m]	67
Fuselage external diameter [m]	5

Fuel volume (total) [m3]	370
Rear tank volume [m3]	290
Underfloor tank volume [m3]	80
Wing loading (at gross mass) [kg/m2]	425
Required thrust-to-weight ratio in subsonic regime	0.22
Required thrust-to-weight ratio in supersonic regime	0.38
Aerodynamic efficiency at Mach 1.5	5.3

The Mach 1.5 case study is conceived to carry more passengers on the same route, flying at a slower speed. It is particularly interesting to see how the method reaches convergence on a smaller empty weight mainly because of the reduced fuel storage needs. The aircraft is also smaller with reference to the Mach 2 case study, featuring a shorter wing/fuselage assembly, even with a larger cabin. This also induces a slightly higher efficiency (with benefit on fuel consumption), with similar wing loading.

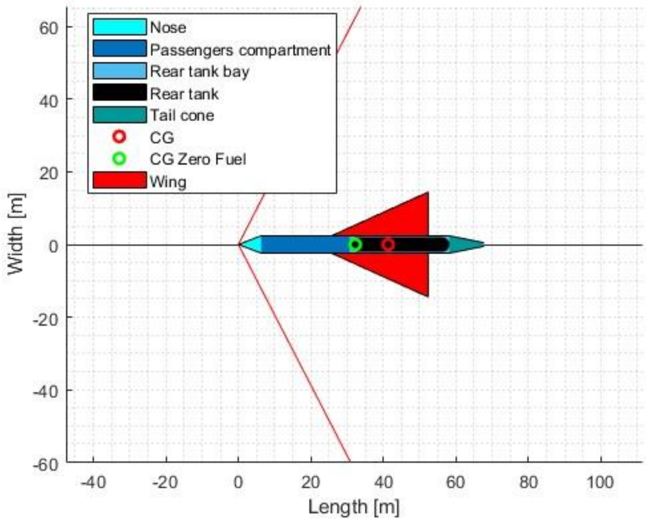

Figure 7 - Vehicle layout, plan view (Mach 1.5 case study)

Thrust-to-weight ratio is similar to the reference aircraft, with very close values of thrust both in subsonic and in supersonic regimes, but particularly showing a higher value in take-off (because of increase of take-off mass) and lower value in supersonic regime (since the aerodynamic efficiency is slightly higher).

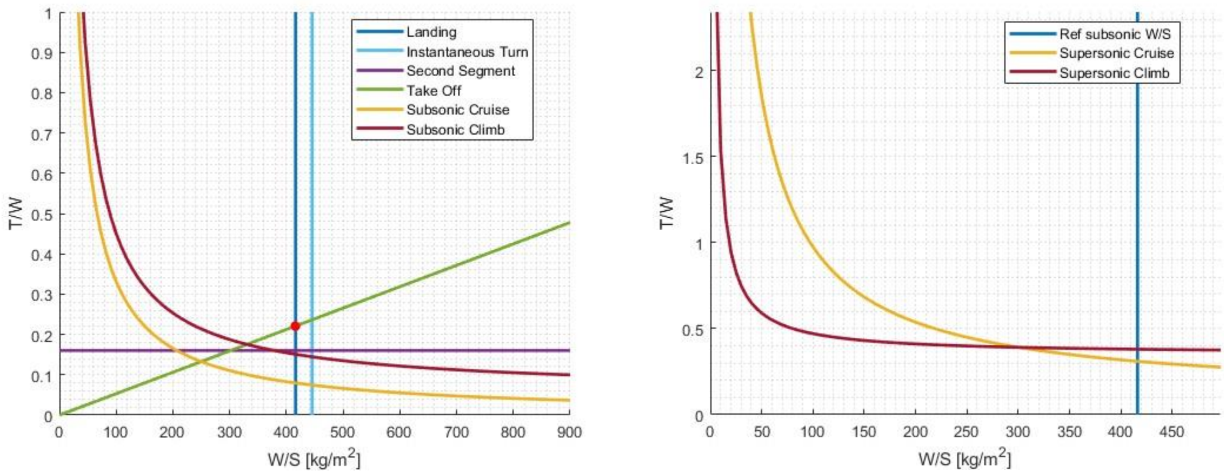

Figure 8 - Matching charts for the Mach 1.5 case study (left - subsonic regime, right – supersonic regime)

Results for the second concept (Mach 2.5) are instead shown in Table 6. In this case, payload requirement is reduced in order to balance the higher amount of fuel required for the faster cruise. However, the increase of volume required to host the extra fuel required for the Mach 2.5 flight has a considerable impact on aircraft configuration, which is heavier and blunter. The vehicle reaches in fact a length of 77 m, with a required fuel volume of around 455 m3. The thrust-to-drag penalty together with a reduced aerodynamic efficiency are affecting the overall concept which results to be more demanding to operate, even with the main configuration values associated to mass, wing loading and thrust-to-weight ratios are still technically in the expected range. Required thrust matching is in fact still similar to the other configurations, but the absolute value of 168 kN per engine are required in supersonic climb.

Table 6 - Main output for Mach 2.5 case study

Variable	**Value**
Operating Empty Mass [kg]	104500
Fuel Mass [kg]	32100
Gross Take-Off Mass [kg]	156000
Wing surface [m2]	400
Wingspan [m]	30
Vehicle length [m]	77
Fuselage external diameter [m]	5
Fuel volume (total) [m3]	455
Rear tank volume [m3]	410
Underfloor tank volume [m3]	50
Wing loading (at gross mass) [kg/m2]	390
Required thrust-to-weight ratio in subsonic regime	0.23
Required thrust-to-weight ratio in supersonic regime	0.44
Aerodynamic efficiency at Mach 2.5	4.8

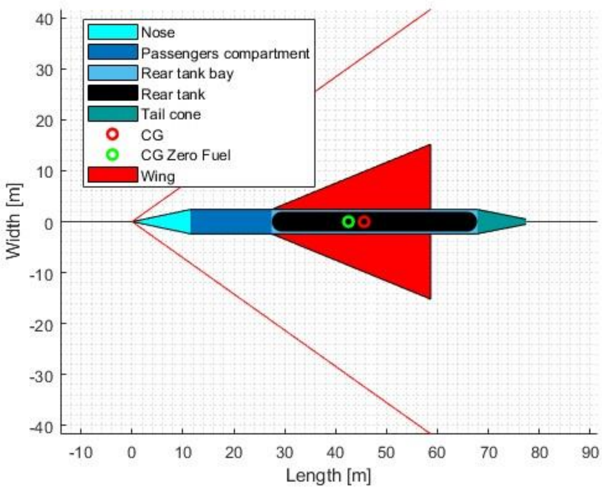

Figure 9 – Vehicle layout, plan view (Mach 2.5 case study)

As final remark it is thus possible to state that the impact of fuel storage needs is deeply influencing the layout of the aircraft configuration, even in presence of similar payload-range requirements, when the Mach number of the supersonic regimes is modified to move towards higher speed regimes. Small modification of speed requirements may lead to non negligible increase of aircraft dimensions and mass, with reduced slenderness parameters and higher fuselage fineness ratio. Aerodynamic efficiency can be also impacted in a non negligible way, even if the aircraft is supposed to have a reduced take-off mass because of the lower fuel quantity (in kg) required. A crucial outcome of the analysis is the composition of the mass breakdown of the aircraft, which tends to be more complex to estimate because of the increase of empty mass due to the presence of dedicated tanks for fuel storage purposes.

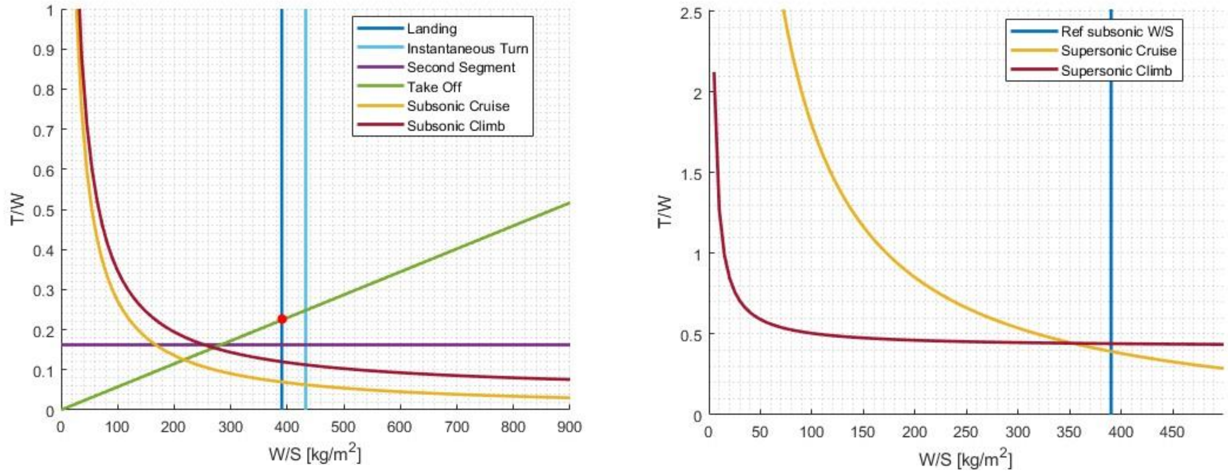

Figure 10 - Matching charts for the Mach 2.5 case study (left - subsonic regime, right – supersonic regime)

The impact of high-level requirements associated to speed appears in fact to produce consequences on the configuration of the aircraft that are usually hidden or non present for traditional kerosene-fuelled vehicles. Notably, the results obtained with the method suggest that more efficient concepts, featuring Concorde configuration and conceived to host cryogenic fuels, such as liquid hydrogen, can be found in the very low supersonic regime, with reasonable payload capacity. As speed increases, the benefit of hydrogen in terms of consumption and environmental impact cannot bare the disadvantages of vehicle growth, with detrimental impact on the operational concept.

5. Conclusions

This paper aimed at evaluating the possibility of designing a liquid hydrogen-powered aircraft for passengers transportation, flying in low supersonic regime along trans-Atlantic routes, with particular focus on the feasibility of Concorde-like configurations modified to host liquid hydrogen on-board. A specific conceptual design methodology was developed, looking carefully at the impact of hydrogen storage needs on vehicle configuration. A reference Mach 2 case study, conceived to fly over 6000 km route with 80 passengers on board was designed, and variation of high level requirements on aircraft configuration were assessed in the range Mach 1.5 – Mach 2.5 and 60 – 100 passengers, coupling case studies having lower speed with higher payload and vice-versa. Results showed that more efficient configurations in terms of thrust-to-drag balance and aircraft dimensions are to be found in the very low supersonic regime, because of the considerable impact of speed on fuel consumption and, as consequence, on aircraft size. The increase in aircraft dimension is in fact detrimental for the overall operational concept, producing configurations which are larger than Concorde architecture even with smaller payload. This also suggests that, for higher speeds, Sustainable Aviation Fuels (SAF) can be a better option, together with the adoption of different configurations for the airframe.

In order to further enhance the model and to validate the results, updates of the method are expected, especially in terms of aerodynamic and propulsive characterization algorithms, as future works. Moreover, the adoption of proper mission simulation campaigns, as a tool to validate these conceptual design results, is envisaged, with focus on the prediction of fuel consumption for different concepts and fuel used.

Acknowledgment

This work has been carried out within the frame of “MDO and REgulations for Low boom and Environmentally Sustainable Supersonic aviation” (MORE&LESS) project. This project has received funding from the European Union’s Horizon 2020 research and innovation program under grant agreement No 101006856.

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Copyright Statement

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