

Article

Potential of Hydrogen Fuel Cell Aircraft for Commercial Applications with Advanced Airframe and Propulsion Technologies

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Abstract: The present work demonstrates a comparative study of hydrogen fuel cells and combustion aircraft to investigate the potential of fuel cells as a visionary propulsion system for radically more sustainable medium- to long-range commercial aircraft. The study, which considered future airframe and propulsion technologies under the Se2A project, was conducted to quantify potential emissions and costs associated with such aircraft and to determine the benefits and drawbacks of each energy system option for different market segments. Future technologies considered in the present work include laminar flow control, active load alleviation, new materials and structures, ultra-high bypass ratio turbofan engines, more efficient thermal management systems, and superconducting electric motors. A multi-fidelity initial sizing framework with coupled constraint and mission analysis blocks was used for parametric airplane sizing and calculations of all necessary characteristics. Analyses performed for three reference aircraft of different sizes and ranges concluded that fuel-cell aircraft could have operating cost increases in the order of 30% compared to hydrogen combustion configurations and were caused by substantial weight and fuel burn increases. In-flight changes in emissions of fuel cell configurations at high altitudes were progressively reduced from medium-range to long-range segments from being similar to hydrogen combustion for medium-range to 24% for large long-range aircraft, although fuel cell aircraft consume 22–30% more fuel than combustion aircraft. Results demonstrate a positive environmental impact of fuel cell propulsion for long-range applications, the possibilities of being a more emission-universal solution, if desired optimistic technology performance metrics are satisfied. The study also demonstrates progressively increasing technology requirements for larger aircraft, making the long-range application's feasibility more challenging. Therefore, substantial development of fuel cell technologies for long-range aircraft is imperative. The article also emphasizes the importance of airframe and propulsion technologies and the necessity of green hydrogen production to achieve desired emissions.

Keywords: hydrogen aircraft; emissions; hydrogen combustion; fuel cells; aircraft design



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1. Introduction

The aviation industry, along with the transportation sector, in general, has been experiencing increasing pressure to improve energy efficiency and reduce emissions in recent

years. To motivate further developments in this direction and achieve acceptable emission levels for sustainable transportation, the European Union, together with representatives from the industry, has set a desired emission target of climate-neutral air mobility by 2050, which is based on a principle of net-zero emissions. The net-zero principle means that the amount of greenhouse gas released into the atmosphere is neutralized [1]. The document also emphasizes the need for reductions in both CO₂ and non-CO₂ emissions.

While aircraft emissions depend on a variety of factors, like flight altitude, atmospheric conditions, airline operations, and air traffic management, which can all be optimized, significant improvements in aircraft performance itself will also be necessary. Two approaches for increasing aircraft efficiency and reducing emissions are possible. Improvements in existing airframe technologies and the introduction of new ones can reduce aircraft weight and improve aerodynamic efficiency, which will make an aircraft more fuel-efficient. Improvements in materials, achieving more laminar airframe, and introducing more efficient solutions for load alleviation are some of the common research directions [2].

Another approach to achieving desired emissions levels is to introduce alternative energy sources and propulsion system technologies. Several solutions are currently being investigated in academia and industry. The use of sustainable aviation fuels (SAFs) is the closest solution to conventional kerosene combustion, as little changes to propulsion technology are required. However, while this technology can theoretically achieve net-zero CO₂ emissions, the problems of NO_x emissions and contrails remain, which are of similar importance to climate impact [3]. Additionally, SAF, if based on bio-fuels, can also bring challenges regarding land usage, water supply, food prices, etc., ref. [4].

Battery-electric or hybrid-battery-electric propulsion systems for aviation are also under investigation. While battery-electric flight has already been realized for light aircraft [5], the key challenge for this technology in commercial aviation remains the poor specific energy of batteries. With state-of-the-art battery technology reaching about 300 Wh/kg, the weight of the batteries considerably limits aircraft size and flight duration [6]. Even with expected improvements in battery technology in coming years and the introduction of advanced airframe technologies for more efficient aircraft performance, battery-electric propulsion, especially for large aircraft, remains challenging [7].

Another promising field of research is hydrogen as a fuel for aircraft propulsion. Here, two main options can be considered: hydrogen combustion in conventional gas-turbine-based engines like turboprop or turbofan or hydrogen fuel cells to supply electric motors. The combustion solution offers the advantage of a potentially easier adaption to existing aircraft concepts [8]. While in-flight CO₂ emissions can be avoided completely, the problems of NO_x emissions and contrails can not be mitigated by utilizing this technology. Moreover, the generic design problem of hydrogen aircraft due to the volumetric energy density [8] of hydrogen and operational safety [9–11] remains challenging.

The use of hydrogen fuel cells eliminates the need for fuel combustion, so NO_x emissions can also be avoided. Thus, regarding emissions, fuel cells could be promising. However, proton exchange membrane (PEM) fuel cells achieve specific powers of not more than 3.5 kW/kg per stack with state-of-the-art technology [12]. Other sources report that current stack power densities of 2 kW/kg were eventually reduced to 1.6 kW/kg due to passive components [13]. Forecasts of fuel cell stack power densities range from 5.5 kW/kg for low-temperature fuel cells [14] to 11 kW/kg by 2035 for low-temperature and 16 kW/kg for high-temperature fuel cells by 2050 [15]. Additional balance of plant components like compressors, humidifiers, and heat exchangers for proper cooling are also required to ensure that the fuel cells operate in their most favorable conditions, and their degradation is minimized. However additional components make the fuel cell system even more complex and heavier. Moreover, heat exchangers can significantly contribute to additional aircraft

drag depending on aircraft size and cooling power requirements. For conventional large aircraft, liquid cooling heat exchangers may not be feasible due to the excessive core drag generated. Studies of Gollnow [16] and Kožulović [17] demonstrated that conventional heat exchangers significantly contribute to the overall aircraft mass and drag, making the fuel cell technology infeasible for medium-range aircraft. Therefore, novel solutions are required to ensure that the heat exchanger drag can be minimized. The latest work of Kösters [18] suggested a potential solution and numerically demonstrated the benefits of a phase-change heat pump (PCHP) cooling strategy for a medium-range aircraft with a hybrid propulsion system powered by fuel cells and a conventional gas turbine.

Observing the overall emissions of the different aviation sectors, the largest amount corresponds to medium- and long-range flights [19]. Based on current potential options for alternative energy sources, hydrogen fuel cells could provide the largest amount of emissions reduction with a potentially achievable fuel cell system performance level. Although fuel cells are currently being used in several flying prototypes, and extensive research on commuter and regional fuel cell aircraft is being conducted, currently available fuel cell technologies are still limited, and even potential forecasts of their improvements do not guarantee that the fuel cells could be better than hydrogen combustion. Previous fuel cell research on medium- and long-range aircraft indicated that mid-range aircraft could potentially be beneficial using advanced fuel cell technologies [14] compared to conventional airplanes, while long-range airplanes required implementations of both future propulsion and airframe technologies [20]. However, comprehensive comparisons between hydrogen fuel cells and combustion aircraft in such segments have not been found. Therefore, a quantitative investigation of the benefits of fuel cells compared to hydrogen combustion aircraft with the presence of potential advancements in airframe and propulsion technologies may provide future guidelines for further developments and quantify possible emission reductions of the fuel cell configuration for more challenging medium- and long-range aircraft segments.

The objective of the present work is to compare the potential emissions and cost effects of hydrogen fuel cell and combustion aircraft concepts for medium- and long-range segments considering future airframe and propulsion technology levels of a 2050 forecast. The comparative study aims to determine the dependencies of costs and emissions on aircraft size for fuel cell aircraft and determine the most promising market segments for future research and development. Therefore, this work will first discuss potential future airframe, propulsion, and energy technologies that could enable commercial aviation applications of hydrogen fuel cell propulsion and summarize assumptions used during the design process. After that, the aircraft sizing methodology, and a validation of the used methodology based on existing work are explained. Finally, the obtained results are discussed from the overall design perspective.

2. Enabling Technologies for More Sustainable Aviation

Before analyzing the potential of hydrogen fuel cell aircraft in the context of commercial aircraft operating for medium- and long-range distances for potential future aircraft of 2050, it is necessary to describe technologies proposed for the next-generation of more sustainable aircraft. This section summarizes the airframe technologies being investigated under the SE2A—Sustainable and Energy-Efficient Aviation cluster project—and describes important assumptions and modeling methods used in the present work.

2.1. Airframe Technologies

Several critical airframe technologies that have been studied previously and are currently being investigated are considered in the present work. The study considers hybrid

laminar flow control (HLFC) for lifting surfaces, advanced load alleviation technologies, and advanced materials for airframe design. Such technologies were initially considered in ref. [21], and preliminary comparative studies indicated that HLFC technology plays the most important role from the airframe technology perspective while other technologies can further improve performance and create a more synergistic effect [7]. The present subsection briefly describes each technology and formulates assumptions or modeling methods for each of them.

2.1.1. Laminar Flow Control

Achieving laminar flow on commercial aircraft remains a challenging task, although extensive theoretical, numerical, and experimental research has been conducted over an extensive period of time. The concept of friction drag reduction can substantially reduce aircraft fuel consumption leading to lower fuel burn and consequently lower emissions. Laminar flow can be achieved using aerodynamic shaping reaching natural laminar flow (NLF) or using active flow control via boundary layer suction. For high-speed aircraft, the concept of hybrid laminar flow control (HLFC) is of particular interest due to the high operating Reynolds number and high wing sweep angle, which triggers cross-flow instabilities, so the implementation of NLF becomes limited. Therefore, to further extend the laminar boundary layer, suction at the leading edge is applied, as shown in Figure 1, to mitigate initial instabilities and extend the laminar flow as much as possible and, therefore, reduce airplane friction drag. The method was studied numerically [22–24] and experimentally [25,26], and several flight test studies [27,28] were performed to demonstrate the potential of the given technology. Moreover, potential applications of the technology at the early conceptual design level demonstrated the potential for fuel burn reduction and estimated the potential technology performance variation along the flight mission [29–31]. Although extensive studies have been previously conducted, challenges related to systems integration in the presence of de-icing and leading-edge high-lift systems, manufacturing, and maintenance of the suction skin and wing surface skin; however, related costs still prevent the technology from being used in current commercial aircraft.

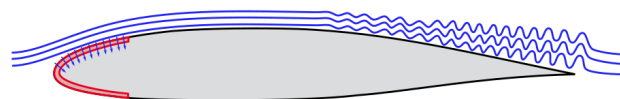


Figure 1. A schematic representation of HLFC technology.

At such an early conceptual design stage, it is unclear how exactly the suction system will be integrated along with the de-icing system. Previous experimental activities [27] showed that both systems can be successfully integrated, so a technically feasible solution exists.

For the present conceptual design investigation, a simplified technique was implemented for aircraft wings to model HLFC effects and size the aircraft. To estimate the effect of laminar flow, the wing was split into several segments along its span. Based on Figure 2, for each segment along the wingspan, the transition Reynolds number Re_T is defined by

$$\frac{Re_T}{10^6} = -4.444 \cdot 10^{-6} \Lambda_{LE}^4 + 3.855 \cdot 10^{-4} \Lambda_{LE}^3 - 1.888 \cdot 10^{-2} \Lambda_{LE}^2 + 3.519 \cdot 10^{-2} \Lambda_{LE} + 29.965 \quad (1)$$

where Λ_{LE} is the wing's leading edge sweep. The transition Reynolds number Re_T is then compared to the current operational Reynolds number at each section along the wingspan, so the transition location becomes

$$X_{tr} = \max\left(\frac{Re_T}{Re}, 0.65\right) \quad (2)$$

The optimistic limit value of 0.65 is selected to ensure that the transition location does not reach unrealistic values. The value of the transition location is then used in the equivalent flat plate model, which estimates the aircraft's parasite drag [32], so the skin friction drag is calculated based on the mixed boundary layer model and directly accounts for the percent laminar flow of the wing. The process of the transition Reynolds number estimation happens at each design iteration, so it accounts for dynamic changes in the planform shape and recalculates the drag coefficient. It must be noted that the method does not account for specific airfoil properties, so transition locations for both upper and lower surfaces are similar. This does not happen in reality, and the transition location highly depends on the airfoil and wing shapes, as well as the flight angle of attack. However, based on studies of Mosca [30], it is possible to design the airfoil with a laminar flow such that differences in chordwise transition locations at cruise, which represents the majority of the flight mission, are minimized. The method is applied to both the wing and the stabilizers (vertical and horizontal).

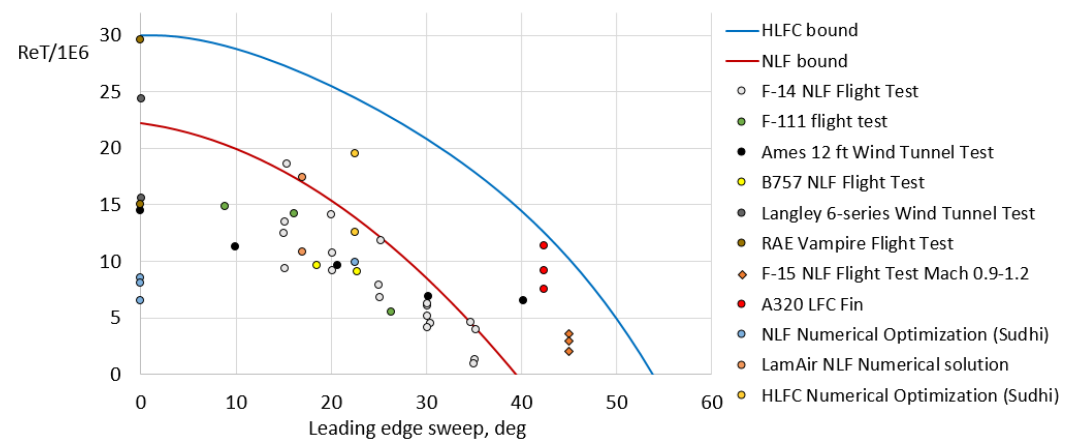


Figure 2. Transition Reynolds number as a function of the wing's leading edge sweep for natural laminar flow and hybrid laminar flow control technologies [33].

2.1.2. Advanced Materials and Structure Concepts

A survey of potential materials and advances in structural design methodologies was performed in ref. [7]. Particularly, potential implementations of hybrid composite materials based on carbon nanotubes and nanofibers [34,35] and tow-steered composites [36] could reduce airframe mass compared to current composite structures. Having a specific strength of up to 300 times compared to high-carbon steels, carbon nanotubes could give an extraordinary combination of tensile strength and elastic modulus with low weight. Tow-steering could provide more optimal composite layups and could optimize the load distribution, therefore making the structure lighter. However, both technologies remain in rather early development stages, and significant manufacturing constraints exist for both solutions.

Given the existing approximations of the effect of advanced materials and forecasts from NASA [37], an average reduction value of 19% of airframe mass with respect to current metallic designs was assumed, corresponding to typical optimistic values given for CFRP materials. At each sizing iteration, the airframe mass is estimated using formulations for a

metallic aircraft. Then, the obtained airframe mass is reduced by the assumed correction factor to estimate the effect of advanced materials.

An additional consideration was taken to estimate the effect of the suction system mass on laminar flow. Kalarikovilagam [38] and Iyer [39] performed preliminary sizing of vertical and horizontal stabilizers of mid-range and long-range commercial jets using the physics-based approach and conducted design trade studies with respect to the number of compressors required for the suction system. Studies showed that horizontal and vertical stabilizers correspond to 0.11% and 0.16% of the empty mass, respectively. The wing suction system mass can be estimated by using the following proportion,

$$\frac{m_{suc,wing}}{S_{suc,wing}} = \frac{m_{suc,empennage}}{S_{suc,empennage}} \quad (3)$$

where $m_{suc,wing}$ and $m_{suc,empennage}$ represent the suction system masses for the wing and the empennage, respectively, while $S_{suc,wing}$ and $S_{suc,empennage}$ are wetted areas where the suction is applied for the wing and the empennage. It is assumed that the engine power off-take strategy is used to power the suction system [39].

2.1.3. Load Alleviation

Although load alleviation techniques have already been used in aviation, more advanced solutions are being investigated. Novel solutions for load alleviation could have active or passive approaches. Passive approaches may include nonlinear stiffness material design [2], viscoelastic damping design, new structural concepts [40], and local morphing structures [2], while active load alleviation methods may include piezoelectric control [41], advanced sensors for more rapid response to gusts [42,43], or fluidic or micro-mechanical flow actuators as a potentially more rapid and responsive method of load alleviation [44]. Load alleviation possibilities were taken into account by reducing the limit load factor from 2.5 obtained using CS 25.337 [45] to 2.0. These numbers assume that load alleviation can counter gust and maneuver loads by at least 20%, while flight safety in the case of load alleviation system failure is retained by assuming safety factors similar to conventional aircraft. A reduced load factor is used in the semi-empirical mass estimation method to obtain a lower wing mass.

2.2. Propulsion and Energy System Technologies

The present study focuses on the comparison of two hydrogen-powered propulsion systems: hydrogen fuel cells and hydrogen combustion. For each propulsion system, a unique set of assumptions and modeling techniques are implemented. The subsection summarizes considerations for each propulsion system.

2.2.1. Hydrogen Combustion Energy Network

The hydrogen combustion energy network has several technological features that are expected to be used in the near future if hydrogen is implemented. From the engine perspective, An ultra-high bypass ratio (UHBPR) turbofan engine is considered as a method to further improve the overall engine efficiency. Its effect on fuel burn has been investigated in several research projects and compared to current engines for medium-range to long-range aircraft [46–49]. A potential reduction of specific fuel consumption (SFC) between 14% and 32% at cruise could be expected for the given technology. For the aircraft sizing in the present work, based on required engine thrust during the design process and available information of three reference engines from the ENOVAL project [49], a proportional SFC reduction percent was applied to the engine model described in Section 3 to account for more efficiency from the UHBPR engine. The proportion is computed based on the

reference engine considered in SUAVE. For each configuration, an engine with a similar pressure and bypass ratio to existing aircraft was mimicked in the sizing framework. Then, an SFC adjustment is applied to achieve the SFC value of similar ENOVAL engines for conventional fuel. This way, the engine is calibrated and only needs hydrogen to be applied. It was also determined that the engine nacelle diameter used for the thrust modeling and drag estimation will be increased by 30% based on the literature study.

Moreover, hydrogen can also be used to cool critical components of the jet engine. The temperature of hydrogen can be efficiently used to cool the turbines and further improve the engine SFC. The present work considers an H₂ expander cycle as a way to provide turbine cooling more effectively. Studies of Brewer [8] suggest that the utilization of such a cycle can further improve the SFC by an additional 4% compared to conventional combustion engines.

2.2.2. Hydrogen Fuel Cell Energy Network

In the hydrogen fuel cell energy network, several components have to be considered. For the fuel cell stacks themselves, a low-temperature proton-exchange membrane (PEM) is chosen, which is assumed to reach a specific power of 5.5 kW/kg [14] at the stack level. The additional balance of plant components includes compressors and humidifiers for the air supply of the fuel cells. A thermal management system (TMS) was included as it has multiple purposes. Predominantly, it needs to act as a cooling system not only for other electric components such as the electric machines, but mainly for the fuel cell stack themselves. Depending on the operating conditions, their efficiency can be as low as 50%, resulting in a large heat flow, potentially in the range of Megawatts, that needs to be dissipated from the aircraft [12]. Proper solutions for heat management are necessary to ensure that fuel cells operate at their favorable conditions and their lifetime is maximized. Large external heat exchangers need to be considered for this task, which can add significant weight and drag to the aircraft. Also, the energy consumption of the cooling system needs to be considered. For this work, the TMS has not been modeled in detail. However, the potential benefits of more advanced TMS systems must be captured. The present work considers the use of PCHP cooling because of its superior performance potential compared to liquid cooling [18]. A more detailed modeling approach would be required to properly account for the PCHP cooling, which may make such early conceptual design studies overly complicated. Instead, a simplified assumption is proposed. Figure 3 shows a comparison of power requirements of two cooling strategies for a medium-range aircraft, featuring a hybrid propulsion system. The comparison shows that the PCHP cooling has significantly less core drag due to the heat exchanger size compared to a conventional liquid cooling option. Because of the overall better performance of the PCHP cooling, its mass and drag are significantly smaller. If all drag power components of the PCHP cooling are added together, then the total drag power is comparable to the mass contribution of a liquid cooling system. Therefore, it is possible to assume that all losses of the PCHP cooling can be covered by estimating the liquid cooling mass instead, so simple equations for mass estimation can be used, and no additional components are required for additional drag estimations. For this purpose, a simplified methodology derived from NASA [50] has been used.

Propulsion is realized by multiple ducted fans driven by high-temperature superconducting electric motors. The aircraft electric power circuit comprises power converters and switches defining the power management and distribution (PMAD) system.

Table 1 summarizes assumptions considered for the components of the fuel cell energy network based either on literature review or expected performance characteristics considered within the project. Although a maximum expected motor power of 6 MW is potentially possible, a deliberately optimistic maximum value of 10 MW is used due to

the excessive power requirements of long-range aircraft. Although such values are overly optimistic, a research and development direction will be recommended in the case of a positive design outcome. Finally, values for an equivalent liquid cooling TMS system are summarized in Table 1 based on the assumption described above.

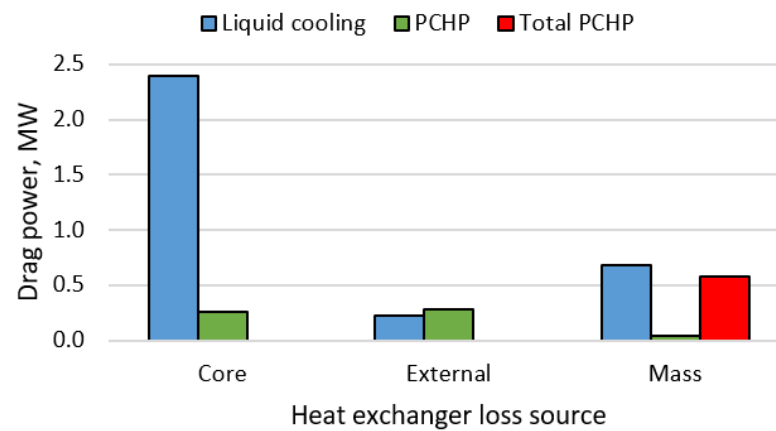


Figure 3. Comparison of liquid and PCHP cooling strategies. Data is based on ref. [18].

Table 1. Energy system technology assumptions.

Technology	Value Range	Units	Reference
Fuel cell stack specific power	5.5	kW/kg	[14]
Fuel cell power density	1.0	kW/L	[51]
Compressor specific power	5.0	kW/kg	assumed
Compressor efficiency	0.85	-	[52]
Heat exchanger design temperature	317	K	[53]
Heat exchanger cooling fluid temperature	380	K	[53]
Maximum expected motor power	6.0	MW	[54]
Maximum desired motor power	10.0	MW	assumed
Motor efficiency	0.99	-	[54]
Motor cooling power loss	10	kW	[55]
PMAD power density	33	kW/kg	[53]
PMAD cooling power loss	10	kW	[53]
Cable density	3.9–5.0	kg/m	[12,53]
Cable efficiency	0.995	-	[53]
Gearbox efficiency	0.98	-	[56]
PMAD efficiency	0.98	-	[53]
Ducted fan pressure ratio	1.5	-	assumed

2.2.3. Liquid Hydrogen Storage

Hydrogen can either be stored as gaseous hydrogen (GH₂) in pressurized tanks or liquid hydrogen (LH₂) in cryogenic tanks. While GH₂ storage may be the easier solution for many applications, LH₂ offers a significant advantage in volumetric storage density, which is of crucial importance for aircraft applications [12]. Even when stored at 164 bar and 288.15 K, GH₂ still has a 5.6 times higher specific volume compared to LH₂ [57]. Thus, cryogenic storage tanks are chosen for this project.

When designing the tank walls, the operating pressure of the system needs to be considered, as well as sufficient thermal insulation to prevent excessive boil-off of hydrogen from the tanks. Winnefeld et al. [57] have shown that the geometrical shape of the tanks can have a great impact on the resulting storage density. Under optimal conditions, and at an assumed operating pressure of about 1.45 bar, it can reach up to 0.64, meaning 0.64 kg of LH₂ per kg of total tank mass. This corresponds to a gravimetric energy density of about

21 kWh/kg, while the volumetric energy density of about 2.2 kWh/L is still about three times lower compared to conventional fuels. The sizing of LH₂ tanks in the present work is based on a simplified heat transfer model, which accounts for the insulation material, geometric properties of the tank, and operating atmospheric conditions. The following section describes the methodology in more detail.

In addition to the tank itself, a heat exchanger needs to be considered to provide the necessary vaporization enthalpy to convert the LH₂ to gaseous hydrogen for the fuel cell at operating pressure. It is fed by waste heat from the fuel cell system itself via the cooling system. The heat exchangers are considered to be integrated into the tanks so that hydrogen can be directly supplied in gaseous form from the tanks to the fuel cell system. Due to the preliminary nature of the study, details regarding the heat exchanger architecture and additional mass penalty are not considered.

3. Aircraft Sizing Methodology

After establishing proper assumptions and methodologies for future airframe and propulsion technologies, a comprehensive conceptual design framework needs to be created to size all proposed aircraft and perform necessary trade and comparative studies.

The initial sizing framework used for the assessment of hydrogen aircraft is shown in Figure 4. The sizing process is divided into two main steps: the initial sizing block and the analysis block. The initial sizing block uses the constraint analysis methodology with the mission analysis to dynamically size the aircraft and explore the design space at early design stages. After sufficient information about the design space is gained and a design point is selected, the concept is manually refined in the analysis block in which additional aspects unavailable in the constraint analysis block can be analyzed. The present section briefly summarizes the methodology, which is described in more detail in ref. [7]. It also focuses on specific features implemented to answer questions of the present work.

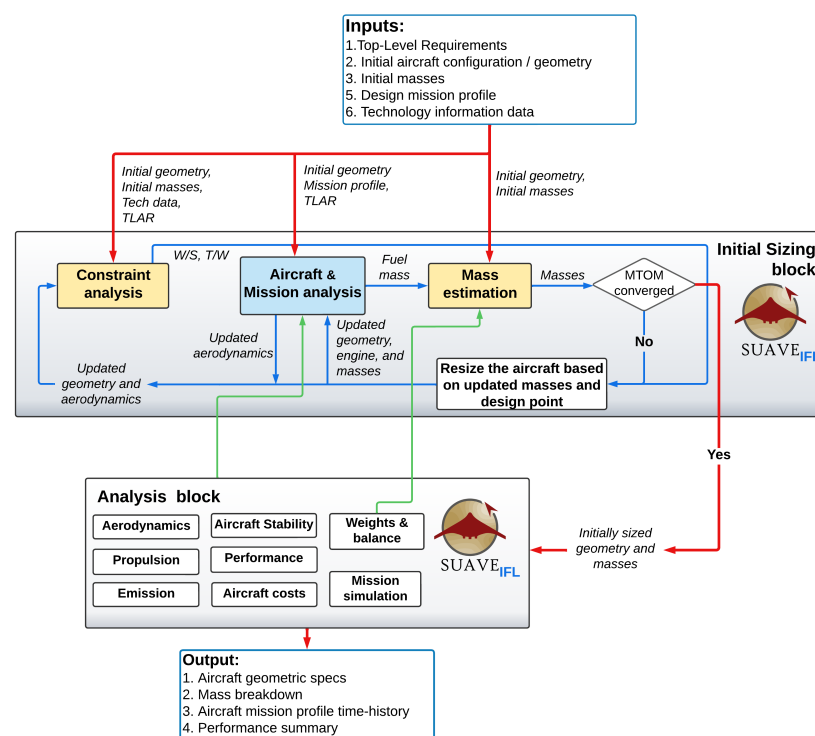


Figure 4. Initial aircraft sizing framework based on SUAVE [32].

3.1. Initial Sizing Framework

Figure 4 describes the initial sizing framework used to design both hydrogen combustion and fuel cell aircraft. Given the aircraft configuration, a geometric parametric modeling technique, the aircraft's initial masses, and technology assumptions, the constraint diagram is initiated. The constraint diagram helps identify acceptable thrust-to-weight ratio and wing loading that satisfy all necessary performance requirements. Information about the concept of constraint analysis can be found in ref. [58]. The initially proposed configuration is used to perform the mission analysis from which the required mission fuel is computed. Then, the aircraft mass breakdown is calculated using the appropriate estimation methods described below. The resulting value of the maximum takeoff mass (MTOM) is compared to the initial guess to calculate the relative error. If a tolerance value is achieved, then the analysis is finished and the configuration is sent to the analysis block. Otherwise, the parametric configuration model and the values of thrust-to-weight ratio T/W and wing loading W/S are used to resize the aircraft based on the geometric model defined by the designer. Finally, information obtained during the mission analysis and new geometric properties of the aircraft are imported back to the constraint analysis module and new masses are imported into the mission analysis module for the next sizing iteration.

3.1.1. Geometric Parameterization

The present work considers configurations schematically shown in Figure 5. Two important aspects unique for each configuration were considered for the present work: the allocation of fuel cells and the number of engines. Hydrogen fuel tanks are allocated above the cabin in both cases, as shown in Figure 6. Although such configuration does not lead to a more aerodynamically efficient aircraft, compared to the configuration with fuselage-level integrated tanks [59], such allocations provide more safety in the case of a crash and help distribute the weight more uniformly without interfering with the cabin. Particularly, a reduced fuselage fineness ratio will increase aircraft pressure drag compared to the fuselage-level integrated option. On the other hand, large aircraft with fuselage-level integrated tanks will require fuel tanks allocated aft and in front of the cabin, which may lead to either a separation of a cockpit from the cabin or the creation of a “catwalk” that would increase the tank mass [60]. Finally, critical damage to over-the-cabin tanks can be lowered due to the presence of the cabin and the baggage compartment below. Therefore, the over-the-cabin option was chosen as a more convenient option from an operating perspective. Fuel cells and supporting systems were assumed to be allocated inside the inner wing between front and rear spars and in the fuselage's extended belly. It was assumed that it could be accessible, and the belly is used only if not enough volume inside the inner wing is available. A given assumption is rather optimistic and requires additional studies to determine if a more conservative allocation of all fuel cell stacks inside a large belly is a more favorable solution.

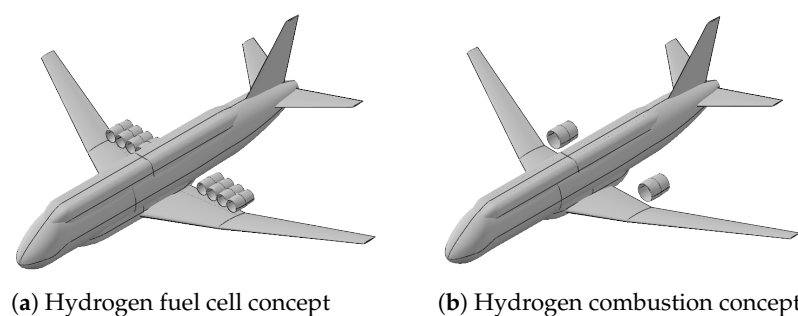


Figure 5. Hydrogen fuel cell and combustion long-range aircraft configurations considered.

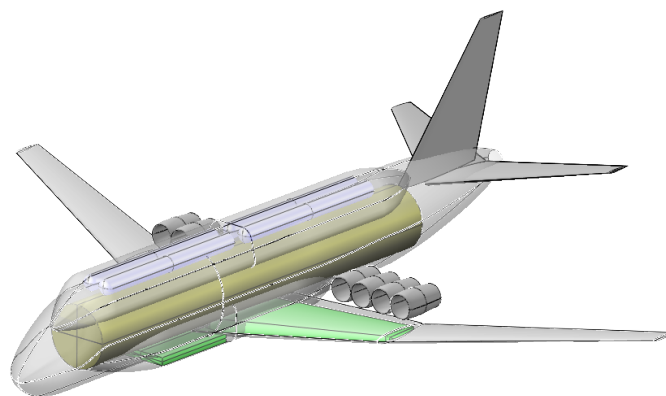


Figure 6. Schematic representation of fuel tanks and fuel cell allocations. Green areas represent fuel cell stacks, blue tanks describe fuel tanks, and yellow represents the cabin.

Fuselage Geometric Sizing

Since an over-the-cabin tank allocation is considered, appropriate methodologies should be developed to automatically size suggested airplanes.

To size the aircraft, information about cross-sectional properties and wetted areas is required to properly estimate the aircraft's drag and fuselage mass. Since the dynamic sizing updates the amount of fuel at every iteration and affects the fuel tank volume, the fuselage shape will also change dynamically. In the case of the over-the-cabin tank allocation, the fuselage cross-section can be split into three segments, as shown in Figure 7, and the total cross-sectional area S_{cc} can be expressed as,

$$S_{cc} = 2(S_1 + S_2 + S_3) \quad (4)$$

Segment S_1 always remains constant and represents the lower half of the fuselage circle (assuming that the pressurized cabin is close to a perfect circle). However, the fuselage diameter also accounts for the piping that will be used to deliver the fuel to the engines. A fixed value of 10 cm was derived from the study of a long-range hydrogen aircraft by Brewer [8] and was used to account for the presence of the piping. The upper cabin portion and the fuel tank segment may change depending on the fuel tank size and may have the following scenarios:

1. If the fuel tank diameter D_t , which includes the actual tank diameter and an extra 10 cm of piping, is less or equal to the cabin radius R_{cab} , then the second segment is blended with the tank segment and makes either a trapezoid or a rectangle depending on the fuel tank diameter. In this trapezoid, the upper side is always equal to the tank diameter D_t and the lower side equals R_{cab} . So, the total cross-sectional area becomes,

$$S_{cc} = (R_{cab} + D_t)^2 + \frac{\pi R_{cab}^2}{2} \quad (5)$$

2. If the tank diameter is greater than the cabin radius, then the upper cabin segment becomes a trapezoid, and the fuel tank segment is defined as a square. The cross-sectional area is then defined by,

$$S_{cc} = 2 \left[D_t^2 + \frac{D_t + R_{cab}}{2} R_{cab} + \frac{\pi R_{cab}^2}{4} \right] \quad (6)$$

The fuselage's total height is defined by

$$H_{fuse} = 2R_{cab} + D_t \quad (7)$$

The effective diameter useful for the form factor estimation used for low-fidelity parasite drag calculations is defined by

$$D_{eff} = \sqrt{\frac{4S_{cc}}{\pi}} \quad (8)$$

The wetted area used for the parasite drag estimation is approximated using the formulation of Raymer [61]:

$$S_{wet} = \frac{3.4}{2}(S_{side} + S_{top}) \quad (9)$$

where S_{side} and S_{top} are the side and top projected areas, respectively. Both areas are dynamically updated at every iteration based on the sizing of the cross-sectional area.

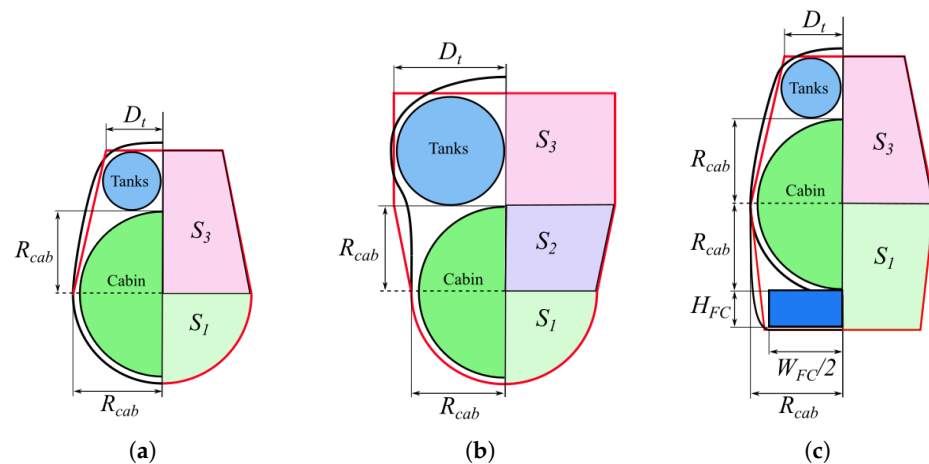


Figure 7. Fuselage cross-section options during the initial sizing process. The black outer line shows the actual shape, and the red one represents the simplified geometry. (a) $D_{tank} \leq R_{fuse}$. (b) $D_{tank} > R_{fuse}$. (c) $D_{tank} \leq R_{fuse}$ with fuel cells.

A special case of fuselage-embedded fuel cells is shown in Figure 7 for a sample case with smaller hydrogen tanks. Generally, the parameterization can be applied for both fuselage configurations with smaller and larger tanks. The fuselage cross-section is extended downward by the height H_{FC} that depends on the size of required fuel cells. The fuel cell width W_{FC} is defined by the designer. Therefore, the lower cabin segment transforms into a trapezoid section and the total cross-sectional area becomes,

$$S_{cc} = \begin{cases} 2 \left[\frac{R_{cab} + D_t}{2} (R_{cab} + D_t) + \frac{R_{cab} + 0.5W_{FC}}{2} (R_{cab} + H_{FC}) \right], & D_t \leq R_{cab} \\ 2 \left[D_t^2 + \frac{D_t + R_{cab}}{2} R_{cab} + \frac{R_{cab} + 0.5W_{FC}}{2} (R_{cab} + H_{FC}) \right], & D_t > R_{cab} \end{cases} \quad (10)$$

Then, the fuselage maximum height is updated according to Figure 7, and the side and top-projected areas are updated based on changes in the fuselage maximum height and width.

Wing and Empennage Geometric Sizing

A sample wing planform that features a kink at an arbitrary trailing edge sweep angle is shown in Figure 8.

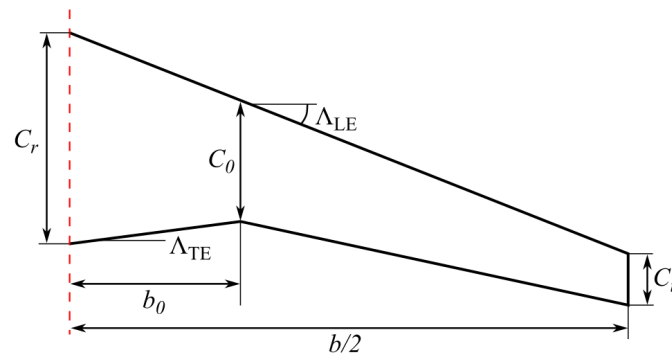


Figure 8. Wing planform with the inboard kink. Adopted from [7].

The total planform area is defined by splitting the wing into the inboard segment with the kink and the outboard segment without the kink and calculating the corresponding areas of two trapezoidal segments

$$S_{wing} = S_{inb} + S_{outb} = \frac{b}{2} [C_r k_b + C_0 + C_t - C_t k_b] \quad (11)$$

Given the wing planform area available from the constraint analysis, the wing root chord is calculated and is defined by

$$C_r = \frac{2 \left(S_{wing} + k_b \frac{AR \cdot S_{wing}}{4} (\tan \Lambda_{LE} + \tan \Lambda_{TE}) \right)}{(k_b + \lambda - \lambda k_b + 1) \sqrt{AR \cdot S_{wing}}} \quad (12)$$

where k_b is the kink span ratio. The derivation of Equation (12) is fully described in ref. [21]. Finally, the rest of the geometry can be obtained using the root chord, initially defined aspect and taper ratios, and wing sweeps. Although the general formulation considers an arbitrary trailing edge sweep of the inner wing, the present research assumes that the trailing edge sweep angle at the kink equals zero. Moreover, the kink size depends on the number and size of engines located on the inner wing, as shown in Figure 5. To do this, the ducted fan sizing procedure is coupled with the wing sizing process. For the hydrogen combustion aircraft, a fixed value of 0.3 was used, while an actual value for fuel cell configurations is computed based on the total width required to fit all electric motors.

The empennage was sized using classical straight tapered wing relations with a fixed tail volume ratio recommended by Raymer [61].

3.1.2. Aerodynamic Analysis

The aerodynamic analysis used in the present research is based on a low-fidelity approach. The lift of the aircraft is estimated using the Vortex Lattice Method code AVL [62] available within SUAVE by default [32]. Aircraft drag is based on the component drag breakdown method in which the total drag coefficient is defined by

$$C_D = C_{D_p} + C_{D_i} + C_{D_c} + C_{D_{misc}} \quad (13)$$

where C_{D_p} is the parasite drag estimated using the equivalent flat plate analogy with a mixed boundary layer [32], C_{D_i} is the induced drag defined using the method derived by Nita [63], and C_{D_c} is compressibility drag modeled using the Korn equation with the sweep correction [64]. Finally, $C_{D_{misc}}$ represents the miscellaneous drag that accounts for drag not covered by other components. A default semi-empirical formulation available in SUAVE was used in this study [32]. Finally, drag penalties related to thermal management

systems were not considered because of the TMS system assumption made in the present work and should be addressed in future studies with a more accurate representation of the PCHP cooling strategy.

3.1.3. Ducted Fan Sizing

The ducted fan disk is sized based on a combination of semi-empirical methods described in Mattingly [65] and Masson [66]. Based on the formulation of Mattingly, the fan diameter is defined by

$$D_f = \sqrt{\frac{T^3}{589\sigma HP^2 N_{bl}^{1/3}}} \quad (14)$$

where T is the fan thrust in pounds, N_{bl} is the number of fan blades, and σ is the blade solidity defined by

$$\sigma = \frac{C_{MAC_{bl}} N_{bl}}{2\pi R_{bl}} \quad (15)$$

where $C_{MAC_{bl}}$ is the blade mean aerodynamic chord, and R_{bl} is the blade radius. Combining Equations (14) and (15) and expressing the blade radius in terms of the fan diameter, the fan diameter becomes

$$D_f = \frac{\pi T^3}{589 C_{MAC_{bl}} N_{bl}^{4/3} HP^2 (1 - A_{hub}/A_{fan})} \quad (16)$$

where the factor A_{hub}/A_{fan} is the hub-to-fan area ratio added by the author to correct the diameter since the hub blocks the effective area that generates thrust.

Another method is based on the motor size, which is embedded into the hub. This way, given the required hub diameter and the hub-to-tip ratio of the ducted fan disk, the overall diameter can be found. To estimate the hub diameter, information regarding the motor size is necessary. Motor minimum volume as a function of its rated power is summarized in ref. [66] and the estimated trend line is defined by

$$V_{mot} = 1.7742 \cdot 10^{-5} P_{mot} - 7.5712 \cdot 10^{-3} \quad (17)$$

where P_{mot} is defined in kW. As a simplification, it is assumed that the electric motor is geometrically described as a cylinder with the diameter of D_{mot} and its length equal to its diameter. Therefore, the total motor volume becomes,

$$V_{mot} = \frac{\pi D_{mot}^3}{4} \quad (18)$$

Equation (18) is solved for the motor diameter D_{mot} . Then, given the value of the hub-to-tip diameter ratio, the fan diameter becomes

$$D_{fan} = \frac{1.15 D_{mot}}{k_{tip}} \quad (19)$$

where k_{tip} is the tip-to-hub diameter ratio assumed to be 0.35, and the factor of 1.15 accounts for an extra 15% of the diameter since volumes defined in Equation (17) represent minimum values.

Finally, based on two different estimations, the maximum value is selected as the design choice.

3.2. Aircraft Mass Estimation

The mass estimation of the designed airplanes is performed using several different methods. The airframe mass estimation is performed using the FLOPS [67] method. The effect of advanced materials is applied as a reduction factor for airframe components that are initially computed for a metallic airframe. The suction system mass is added to all standard systems available in the FLOPS method. The wing mass estimation also considers a reduced limit load factor which accounts for the presence of load alleviation technologies. The hydrogen energy network masses are estimated using either semi-empirical or physics-based models described below.

3.2.1. Fuel Cell System Mass Estimation

For the fuel cell system, a simplified model was chosen due to the preliminary nature of the present research and the multi-disciplinary nature of the work. The present formulation of the fuel cell system mass consists of the fuel cells, air compressors, waste heat exchangers, and humidifiers, while other balance of plant components are assumed to be included in the fuel cell systems and not modeled separately. The mass calculations for the system are shown below, while a detailed description of the system model can be found in the Appendix A.

The fuel cell mass is defined by,

$$m_{FC} = N_{stack}(m_{FC} + m_{comp} + m_{cool} + m_{humid}) \quad (20)$$

where m_{FC} is the mass of fuel cells per stack, m_{comp} is the compressor mass, m_{cool} is the cooling system mass, and m_{humid} is the mass of the humidifier. It assumed that all mass components are defined per stack and are then multiplied by the required number of stacks N_{stack} .

The fuel cell mass is defined using the fuel cell power density obtained from the literature, so the mass becomes,

$$m_{FC} = \frac{P_{FC_{max}}}{\rho_{FC}} \quad (21)$$

where $P_{FC_{max}}$ described the maximum fuel cell power of the stack sized to operate at all critical flight conditions, and ρ_{FC} is the fuel cell power density.

The compressor mass is defined in a similar fashion using a defined power density, so

$$m_{comp} = \frac{P_{comp_{max}}}{\rho_{comp}} \quad (22)$$

Similar to Equation (21), the compressor mass is calculated using the maximum required compressor power $P_{comp_{max}}$ and the compressor power density ρ_{comp} .

The present work considers a liquid cooling strategy similar to the one described in ref. [50], in which a simplified model was derived using the heat exchange circuit analysis approach, and simple equations were derived based on sample loads and optimizations of a steady-state analysis of the model. The cooling system mass is described by

$$m_{cool} = (0.194P_{heat,reject} + 1.39)f(dT) \quad (23)$$

where $P_{heat,reject}$ is defined per stack, and $f(dT)$ is described using Equation (A11).

Finally, the mass of the humidifier was obtained from ref. [12] and is defined by

$$m_{humid} = 1.3669 \ln P_{FC_{max}} + 0.2644 \quad (24)$$

where the stack fuel cell maximum power $P_{FC_{max}}$ is defined in kW.

3.2.2. Fuel System Mass Estimation

The hydrogen fuel system mass estimation was performed using the approach described in ref. [68]. The present paragraph briefly summarizes the approach, while comprehensive information can be found in ref. [68]. The fuel system was split into several components: fuel tanks, heat exchangers, valves, refueling systems, venting systems, and pipes. Masses of all components except for fuel tanks were described using the survey by Brewer [8] and the summary estimated weights of each component are summarized in Table 2.

Table 2. Masses for each fuel system component. Adopted from [7].

Component	Weight	Units
Boost pumps	53.1	kg/tank
High-pressure pumps	6.0	kg/engine
Pump/valves electrical systems	48.5	kg
Engine fuel delivery lines	0.065	kg/m
Refuel/defuel system		
Refuel lines inside tank	7.2	kg/tank
Valves and adapters	21.3	kg
Refuel/defuel manifold	239.5	kg
Tank vent/pressurization system		
Valves	3.5	kg/tank
Tank pressure generation system	2.5	kg/tank
Vent lines	0.98	kg/m

The hydrogen tank mass estimation is based on structural and thermodynamic analyses. The thermodynamic analysis was performed using the method described by Verstraete [69] and Winnefeld [57]. Given the tank geometry and materials used for its structure and insulation, a heat transfer analysis is performed to determine the internal pressure inside the tank and the hydrogen boil-off rate so the fuel burn can be adjusted, accounting for losses. An equivalent resistance model was used to model the heat transfer from the atmosphere to the fuel and estimate the fuel boil-off. The region of the tank insulation was split into several equal segments, at which the local temperature was computed based on heat transfer equations. Given representations of the heat transfer mechanism for each segment, a system of equations is established,

$$\begin{bmatrix} -1 & 1 & 0 & \cdots & \cdots & \cdots & 0 \\ 0 & -1 & 1 & 0 & \ddots & \ddots & \vdots \\ \vdots & 0 & -1 & 1 & \ddots & \ddots & \vdots \\ \vdots & \ddots & \ddots & \ddots & \ddots & 0 & \vdots \\ \vdots & \ddots & \ddots & \ddots & -1 & 1 & 0 \\ 0 & \cdots & \cdots & \cdots & 0 & -1 & 1 \end{bmatrix} \begin{pmatrix} T_{H_2} \\ T_{wall} \\ T_{ins,N} \\ \vdots \\ T_{ins,2} \\ T_{ins,1} \\ T_{skin} \\ T_{air} \end{pmatrix} - \begin{pmatrix} R_{int} \\ R_{wall} \\ R_{ins,N} \\ \vdots \\ R_{ins,2} \\ R_{ins,1} \\ R_{ext} \end{pmatrix} \dot{Q} = \underline{0} \quad (25)$$

where parameters R represent resistances corresponding to each layer of the tank structure, T represent temperatures at each level, and $\dot{Q}$ represents the heat transfer rate. The system is solved iteratively to then obtain the fuel boil-off rate,

$$m_{boil-off} = 1.3 \frac{\dot{Q}}{h_v} \quad (26)$$

where factor 1.3 corresponds to the heat leak from the piping, support structures, and additional systems [69]. More information regarding the modeling of the tank heat transfer

can be found in ref. [7]. Based on the selection of the insulation material thickness that provides acceptable values of the fuel boil-off, its weight can be calculated based on the material density.

The structural mass of the tank is computed based on the hoop stress formulation for a pressure vessel described by Verstrate [69]. The minimum tank wall thickness is defined by

$$t_{wa} = \frac{p_{des} D_w}{2\sigma_a e_w + 0.8P_{des}} \quad (27)$$

where σ_a is the allowable stress considered as one-fourth of the ultimate material stress [70], e_w is the weld efficiency, and P_{des} is the design pressure computed using recommendations of Brewer [8]. The elliptical minimum thickness is defined by

$$t_h = \frac{P_{des} D_{w,h} K}{2\sigma_a e_w + 2P_{des}(K - 1)} \quad (28)$$

where $D_{w,h}$ is the inside diameter of the hemispherical head and the constant K is defined by

$$K = \frac{1}{6} \left[2 + \left(\frac{D_{w,h}}{D_{1,h}} \right)^2 \right] \quad (29)$$

where $D_{1,h}$ is the minor diameter of the elliptical head. Given the minimum thickness of the tank walls and the material used, the tank weight can be computed using the material density and the structural volume of the tank.

The present work considers cylindrical tanks as a universal option in terms of structural mass and more uniform heat transfer characteristics. The insulation thickness-to-diameter ratio of 15% was used based on trade studies performed during initial sizing iterations that provided the lowest fuel boil-off, while the rohacell foam was used as a more conventional insulation option. A vent pressure of 1.55 bar, fill pressure of 1.2 bar, and minimum allowable pressure of 1.1 bar were used as reference values for hydrogen tank sizing and simulations based on typical values used in Refs. [8,57,69]. The minimum atmospheric pressure for the tank sizing was set to the aircraft service ceiling.

Finally, the mass of a high-temperature superconducting electric motor is calculated using the data provided by Stückl [54]. The motor power-to-weight ratio is defined by

$$\frac{P_{Mot}}{W} = 2.63(P_{Mot})^{0.277} \quad (30)$$

where P_{Mot} is the motor power in kW and the power-to-weight ratio P_{Mot}/W is defined in kW/kg. Knowing the required motor power, the power-to-weight ratio is obtained, which leads to the motor mass.

3.2.3. Estimation of Direct Operating Costs

The direct operating costs (DOC) are computed using the TU Berlin method summarized in ref. [71],

$$DOC = DOC_{En} + DOC_{Crew} + k_{advtech} \cdot DOC_{Ma} + DOC_{Cap} + DOC_{Fees} \quad (31)$$

where DOC_{En} are energy costs, DOC_{Crew} are crew costs, DOC_{Ma} are maintenance costs, and DOC_{Cap} and DOC_{Fees} are capital and fees costs, respectively. All costs presented in this work are referenced to the year 2020. In Equation (31), energy costs DOC_{En} are directly proportional to the fuel burn and fuel price. Labor costs and fees were taken from the reference year of 2010 and were corrected by the inflation factor depending on the year 2050. The factor $k_{advtech}$ was added to maintenance costs as a gain factor to account for the

difficulty of the future technologies' maintenance and was assumed to be 2.0. The airframe price used for the capital costs was estimated using the method of Roskam [72] with all parameters related to future technologies maximized. The LH₂ tank price was also added using the method of Hoelzen [73].

The price of liquid hydrogen was estimated using the survey of Hoelzen [73]. The total price is defined by

$$p_{LH2} = p_{prod} + p_{liq} + p_{trans} + p_{store} + p_{refuel} \quad (32)$$

where the total price consists of production, liquefaction, transportation, storage, and refueling components. The future price is subject to uncertainties and could have various price scenarios. Hoelzen describes three possible scenarios with different ranges: optimistic, base, and pessimistic. For the present work, the base scenario of 4.86 EUR/kg was used.

3.2.4. Emission Modeling

The emission model initially provided by Scholz [3] was used for the present study. The model represents the overall effect on the environment as an equivalent CO₂ emission parameter, which consists of the carbon dioxide effect and additional normalized contributions of other emission components that affect global warming. The original model was provided for kerosene and hydrogen combustion options with an assumption of contrails being dependent only on altitude. The formulation was later extended to account for the fuel burn of specific aircraft [74]. The present research was conducted while the modification of the emissions model was taking place, so an alternative modification of the contrails formulation was developed. The final model of the equivalent CO₂ emission for a generic hydrogen aircraft is defined by

$$m_{CO2,eq,GT_i} = (k_1 k_2 EI_{NO_x} f_{km} CF_{mid,NO_x} + k_3 k_4 k_5 CF_{mid,AIC}) \Delta R_{km} \quad (33)$$

where the NO_x emission index EI_{NO_x} is calculated using the Boeing Method 2 [75]. If the hydrogen fuel cell network is used, then EI_{NO_x} equals zero. ΔR_{km} represents the incremental range in km, f_{km} is the local fuel flow rate in kg per km, and CF_{mid} is the correction factor to convert the value into the equivalent CO₂ emission defined by

$$CF_{midpoint,NO_x} = \frac{SGTP_{O3s,100}}{SGTP_{CO2,100}} s_{O3s} + \frac{SGTP_{O3L,100}}{SGTP_{CO2,100}} s_{O3L} + \frac{SGTP_{CH4,100}}{SGTP_{CO2,100}} s_{CH4} \quad (34)$$

$$CF_{midpoint,AIC} = \frac{SGTP_{contrails,100}}{SGTP_{CO2,100}} s_{contrails} + \frac{SGTP_{cirrus,100}}{SGTP_{CO2,100}} s_{cirrus} \quad (35)$$

where $SGTP$ is the sustained global temperature potential, and s is the forcing factor that increases or decreases the hazardous effect of the compound depending on the altitude. Values of $SGTP$ and s are provided by Dallara [76] and are shown in Table 3 and Figure 9, respectively.

Table 3. SGTP indices. Adapted from [76].

Component	Parameter	Units
CO ₂	3.58×10^{-14}	K/kg _{CO₂}
O _{3s}	7.79×10^{-12}	K/kg _{NO_x}
O _{3L}	-9.14×10^{-13}	K/kg _{NO_x}
CH ₄	-3.90×10^{-12}	K/kg _{NO_x}
Contrails	1.37×10^{-13}	K/km
Cirrus	4.12×10^{-13}	K/km

One of the major uncertainties of the present emissions model is related to its averaged properties. Particularly, the dynamics of contrail formation are highly sensitive to both latitudes and altitudes at which the aircraft flies. Based on aircraft design studies of Barton [77], aircraft fleets may significantly depend on the route if an emissions minimization strategy is employed. For equatorial regions, low-altitude flights are preferred to minimize contrails while high-altitude operations are recommended for mid-latitude regions, as shown in Figure 9. The research will attempt to observe emissions in both equatorial and mid-latitude regions to capture a wide range of possible routes and compare overall emissions for both fuel cell and combustion configurations. Therefore, it is necessary to ensure that the average model would provide comparable forcing factors for contrail formation for desired altitudes that were chosen to be 8000 m for the equatorial region and 12,000 m for the mid-latitude region. Observing the forcing factor diagram for contrails and the correlation of the ice super-saturated regions (ISSRs) shown in Figure 9 for selected latitudes and altitudes, a similar ratio of the forcing factors and the ISSR frequency ratios can be observed. Particularly, assuming that an average line in the ISSR frequency diagram is located between altitudes of interest and taking the ISSR frequency value at the altitude at which the forcing factor equals one, similar ratios of forcing factors and ISSR frequencies with respect to the reference point can be obtained. Therefore, it can be assumed that the averaged model will proportionally scale the contrail emissions according to the ISSR, so some level of reliability is reached.

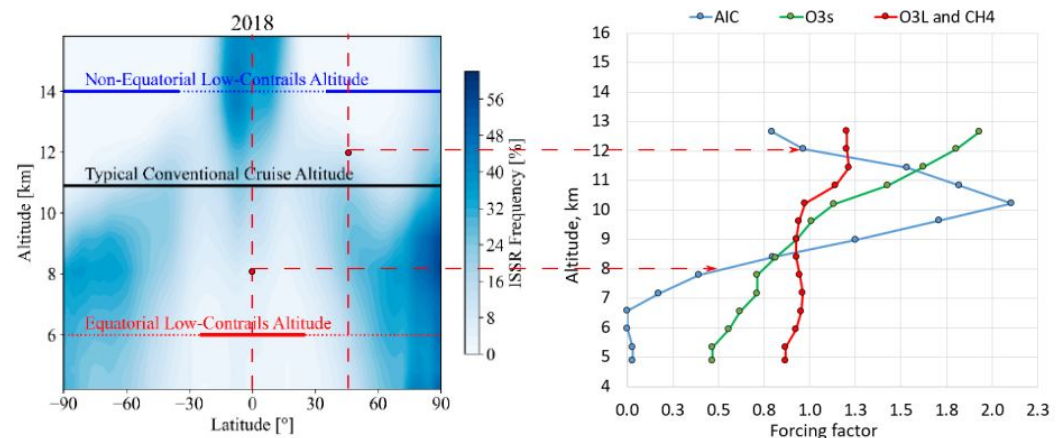


Figure 9. Forcing factor as a function of altitude and its correlation with the possible ice super-saturated regions (ISSRs). Adapted from [76,77].

Correction factors $k_1 - k_5$ in Equation (33) correspond to different effects of the hydrogen fuel and the powerplant system. Typical values of the hydrogen-powered energy network are summarized in Table 4.

Table 4. Correction factors for the equivalent CO₂ emissions for hydrogen combustion propulsion system.

Constant	Value	Influence Description
k_1	2.79	Higher hydrogen combustion temperature
k_2	0.1–0.75	Lean combustion
k_3	Equation (36)	Higher H ₂ O emission
k_4	0.27–0.36	Thinner ice crystals and less visibility of contrails
k_5	Equation (39)	Aircraft size

In the case of hydrogen fuel cells, factors k_1 and k_2 are not used. The factor k_3 for fuel cells can be computed using the relation provided in ref. [52]. Generally, the factor k_3 describes the ratio of water emission indices between kerosene and hydrogen combustion propulsion systems. The water emission ratio of a conventional kerosene-powered gas turbine engine equals 1.23 kg of water per kg of fuel. The water emission factor for hydrogen-powered aircraft is computed by

$$k_3 = \begin{cases} 2.58 & \text{LH}_2 \text{ combustion} \\ \frac{\dot{m}_w}{\dot{m}_f} & \text{LH}_2 \text{ fuel cell} \end{cases} \quad (36)$$

where $\dot{m}_f$ describes the fuel burn, and the water production $\dot{m}_w$ is defined from ref. [52] by

$$\dot{m}_w = 9.34 \cdot 10^{-8} \frac{P_e}{V_c} \quad (37)$$

Finally, the formation of contrails depends not only on the atmospheric properties at which the aircraft flies but also on the optical properties of ice crystals. Studies from Jeßberger [78] showed that the properties of ice crystals and their effects on non-CO₂ emissions depend on the aircraft size. Flight test studies analyzed contrails of A319, A340, and A380 airplanes and measured various important characteristics. Several of those were used to correlate contrails and the aircraft size. The amount of ice crystals created by an aircraft per flight distance is defined by

$$N_{ice} = f_s EI_{soot} \dot{m}_f \quad (38)$$

where f_s is the survival factor assumed to be 0.72, EI_{soot} is the soot emission index, and $\dot{m}_f$ is the fuel burn. This parameter may indirectly correlate with the aircraft size and its tendency to create contrails since more ice is created for larger airplanes. Moreover, the optical depth of ice crystals was observed to be dependent on the aircraft size and also affect the non-CO₂ emissions [78]. Finally, Jeßberger demonstrates an example comparison of imaginary fleets of A319 and A380 and their effects on the contrail formation. If the world aircraft fleet is replaced by A319, the total contrail formation will be 0.73 of the current one while the full A380 fleet provides 2.5 times more contrails. Figure 10 represents all three correlations described above. In the figure, ice crystal amount and optical depth ratios were obtained from the reference and were normalized with respect to the largest value. This information can show dependencies of the given parameters with respect to the aircraft size. Fleet contributions of A319 and A340 were then used as correction factors, assuming that A320, with its fuel burn, would be close to one; this was used as an example reference for the original emissions model.

Given actual contrail formation factors for two aircraft and using tendencies for three airplanes with respect to ice formation and optical depth, linear and quadratic distributions can be used to derive relations for contrail formation factor distribution. However, since it is unknown if a quadratic or a linear relationship shall be used, an average trend was derived as a function of the fuel flow rate. It is important to note that the average value does not approach zero as the fuel flow rate reduces. Therefore, the upper-level quadratic model should be used for fuel flow rates approaching zero. The final correction factor correlation is then defined by

$$k_5 = -0.0063\dot{m}_f^2 + 0.2447\dot{m}_f + 0.2223 \quad (39)$$

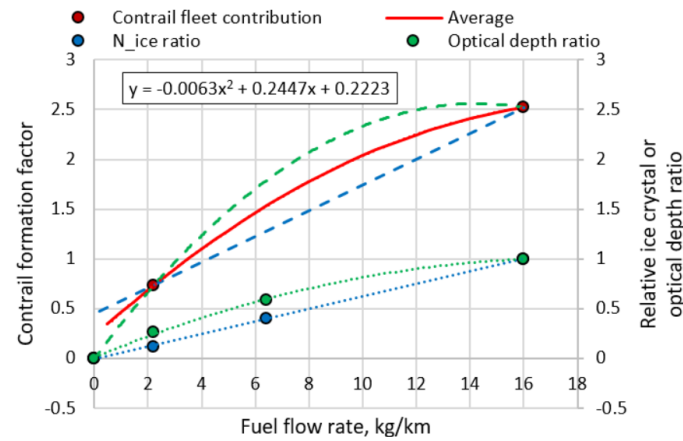


Figure 10. Comparison of correlations of jet engine exhaust.

Finally, the dependence of contrails on the aircraft size does not represent potential differences in water emission for fuel cell and combustion configurations that may also significantly affect the overall outcome. Due to a lack of information, it was assumed that the effect of both propulsion systems was equal.

3.2.5. Dynamic Aircraft Sizing

Having all critical components defined and having initial conditions specified, the framework performs an automatic sizing of both concepts is performed using the following steps:

1. Perform initial fuel cell (if applicable) and propulsor sizing summarized in the appendix based on initial aircraft inputs.
2. Construct the constraint diagram and isolate a design point with specific W/S and T/W or P/W that satisfies all constraints.
3. Estimate empty mass breakdown based on the formulation in Section 3.2.
4. Run the mission and performance analyses and obtain the fuel mass m_f and all necessary performance metrics.
5. Knowing the value of the take-off mass for the present iteration, obtain the reference area using the wing loading value obtained in Step 2.
6. Obtain all wing geometric characteristics based on constants and the S_{wing} using Section “Wing and Empennage Geometric Sizing”.
7. Find the required fuel volume V_f given the fuel mass m_f and liquid hydrogen density.
8. Given the required fuel tank volume and a fixed tank compartment length based on the fuselage length, obtain the required tank inner radius R_f .
9. Given that the insulation is defined outside of the structural layer and the insulation thickness-to-diameter ratio is defined by the designer, the insulation thickness becomes

$$t_{ins} = \frac{t/R_{ins} \cdot R_f}{1 - t/R_{ins}} \quad (40)$$

where t/R_{ins} is the insulation thickness-to-radius ratio. So the tank diameter becomes

$$D_t = 2(R_f + t_{ins} + t_{struct}) \quad (41)$$

where t_{struct} is the structural thickness initially defined by the designer or available from the previous iteration.

10. Recalculate parameters of fuselage cross-section using equations described in the “Fuselage Geometric Sizing” Section and calculate the new fuselage wetted area.

11. Based on updated tank geometric properties, perform the thermodynamic analysis of tanks using the method described in ref. [68]. Obtain boil-off and venting masses.
12. Calculate the new value of the maximum take-off mass.
13. Resize the propulsion system based on the new maximum take-off mass using either formulation for the fuel cell aircraft summarized in Section 3.1.3 and Appendix A or default methods in SUAVE for turbofan engines.
14. Update the constraint diagram inputs of aerodynamic properties based on results obtained during the mission and performance analyses.
15. Compare the new value of MTOM to the old value. If not converged, repeat Steps 2–12.
16. Compute aircraft DOC and equivalent emissions using Sections 3.2.3 and 3.2.4. Compute other performance metrics necessary for comparisons.

3.3. Design Methodology Verification

After establishing the design and analysis framework, it is time to verify if the methodology performs correctly. The model was verified with respect to a reference aircraft presented in the research work of Waddington [14].

3.4. Verification of the Numerical Model with the CHEETA Reference Aircraft

To ensure that the sizing framework operates correctly, verification was performed using the CHEETA reference aircraft shown in Figure 11.

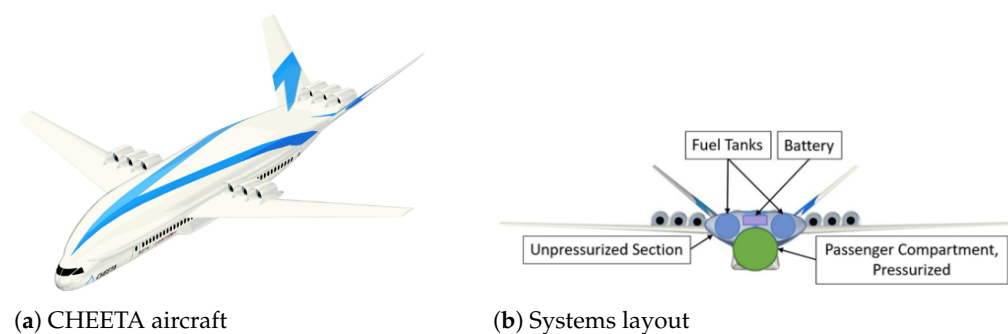


Figure 11. CHEETA reference aircraft layout [14].

The configuration mimics the reference B737-800 aircraft (Boeing, Arlington County, VA, USA) but features the hydrogen fuel cell propulsion system. The aircraft features nine ducted fans powered by superconducting electric motors. Three out of nine engines are located in the aft of the aircraft, while the remaining ones are positioned on the wing. Eight hydrogen fuel tanks are located above the cabin, while fuel cell stacks are positioned near the engines both in the wing and at the aircraft's aft. The set of top-level requirements is summarized in Table 5.

Rather limited information about the final design airplane characteristics and the modeling approach was shown in ref. [14]. In particular, the approach of the ducted fan modeling and sizing, the final sized propulsion system characteristic, and the fuel flow rate formulations were not described. Nevertheless, typical ducted fan propulsion system characteristics were included in the current model. The mission profile similar to the reference B737-800 at the design cruise altitude was used, and the wing loading similar to the one obtained by Waddington was used. The aircraft's wetted area and projected cross-sectional properties were calculated using projected views of the reference aircraft [14].

Table 5. CHEETA concept top-level requirements [14].

Parameter	Value
Passengers (single class)	180
Cabin	Single aisle
Span (at airport gate)	Within 79–118 ft
Takeoff field length	8200 ft
Range	2935 NM at end of life
Cruise Mach	0.78
Cruise altitude	37,000
Reserves	200 NM divert, 30 min loiter, 5% contingency
Regulation	FAR 25 compliant
CO ₂ emissions	0.0 lb/pmi (0.2 lb/pmi for B737-800)
NO ₂ emissions	0.0 lb/LTO cycle (27 lb/LTO cycle for B737-800)

Comparisons of masses and wing geometric properties obtained during the sizing loop are shown in Tables 6 and 7. Good agreement for overall masses and geometric characteristics was achieved with the present model. The total power density of 3.0 kW/kg, including BOP, was obtained in the current model, which is relatively close to the reference value of 2.7 kW/kg demonstrated in ref. [14]. It is important to note that although Figure 11 shows a battery as a part of the propulsion system to deal with transient current loads, its mass was not estimated in the CHEETA aircraft research. Similarly, the present work does not account for the potential battery mass required for a more stable operation of the fuel cell propulsion system.

Table 6. Mass comparison between the CHEETA aircraft and the current model.

Parameter	CHEETA	Current Model	% Difference
Maximum take-off mass, kg	86,400	87,281	1.02
Operating empty mass, kg	62,868	63,867	1.59
Payload mass, kg	15,875	15,875	0.00
Fuel cell system mass, kg	9108	9427	3.50
Fuel tank mass, kg	4976	4843	−2.67
Fuel burn, kg	7655	7582	−0.95

Table 7. Comparison of sized wing geometric characteristics between the CHEETA aircraft and the Current model.

Parameter	CHEETA	Current Model	% Difference
AR	10	10	0.0
Wing area, m ²	171	176	2.9
Kink span ratio	0.39	0.38	2.6
Wingspan, m	41.3	41.9	1.4
Taper ratio	0.3	0.3	0.0

4. Results

The objective of the design space exploration is to answer if fuel cells represent potential emission benefits with comparable cost compared to the hydrogen combustion configuration for high-speed aircraft and for what types of fleet these benefits occur. It is important to note again that the study considers future technologies as a potential enabler of fuel cell propulsion for high-speed aircraft, so a visionary study can be performed without current technology limitations.

To answer the question, three common aircraft types were considered: Airbus A320 (Airbus, Blagnac, France), Boeing B787-9, and Boeing B777-300ER (Boeing). Design range, payload, and takeoff and landing distances were considered similarly to those of the reference aircraft. However, the cruise speed was assumed based on previous studies [68], where it was assumed that airport operations during turnarounds could be modified such that the turnaround time can be reduced by 5%, so normal cruise speed can be reduced proportionally. However, the maximum cruise speed must be similar to or greater than the one typically used for current reference aircraft. Top-level requirements for those three aircraft are summarized in Table 8. Standard mission profiles for each aircraft were considered with reserve requirements specified by CS25 regulations [45]. Benefits of each technology were accounted for by both the main mission and reserves. For instance, it was assumed that the boundary layer suction was utilized for the reserves segment as well.

Table 8. Selected aircraft top-level requirements.

Characteristic	Medium-Range	Long-Range B787-9 Category	Long-Range B777-300ER Category	Units
Maximum payload mass	19.9	53.0	70.0	t
Cruise Mach number	0.75	0.81	0.81	-
Maximum Cruise Mach number	0.78	0.84	0.84	-
Design range	6100	14,140	13,649	km
Takeoff field length (MTOM, ISA)	2100	2900	3100	m
Landing field length (MLM, ISA)	1370	1870	2150	m

For each aircraft category, two airplanes were sized: a hydrogen combustion one and one featuring fuel cells, as shown in Figure 5. The LH₂ combustion configuration features two engines located above the wing similar to the configuration described in ref. [21], while the fuel cell configuration has the number of engines that satisfies the constraint of up to 3 MW per engine for medium-range aircraft and up to 9 MW per engine, as mentioned in Table 1. The design space exploration was performed in two major sizing steps and was related to trade studies for the wing loading ratio, aspect ratio, wing sweep angle, and taper ratio. First, a rather wide design space range was considered, and the second iteration was performed for similar design variables but for a narrower range to refine the design further. For each trade study, sweeps of studies with a range of variables are performed, and then the configuration that features the lowest fuel burn is selected. The determination of the selection criterion was based on the desire to reach the configuration with the lowest emissions. When the most promising configuration is selected, constraints such as takeoff and landing performance and minimum allowable tip chord length (based on reference aircraft) are checked to ensure that critical requirements are satisfied. It is important to note that airport span regulations were deliberately not considered in the present work to explore configurations beyond conventional airport regulations. The only requirement of the maximum wingspan of 80 m was considered an absolute limit based on the ICAO Annex 14 regulations [79]. Based on those optimistic studies, it can be determined how much span constraints may increase the fuel burn and the overall feasibility of each configuration. If the lowest fuel burn configuration violates critical constraints, then the closed configuration that satisfies all constraints is selected. After two major sizing steps were completed, manual refinements to the design were performed to ensure reasonable sizes of stabilizers and applicable CG envelopes.

Both configurations represent tube-and-wing aircraft with over-the-wing mounted engines. For the combustion configuration, classical single-slotted Fowler flaps were considered for the inner and outer wing segments excluding the engine segment. For the fuel cell configuration, it is considered that the inner wing with several ducted fans may have the capability of a Coanda flap to replicate the lift that would occur if fans mimic the lift of slotted flaps in the given segment. Particularly, since ducted fans are attached to the rear spar, a small plan flap can be added to deflect the ducted fan outflow and achieve lift augmentation. The present work does not consider a detailed aerodynamic analysis of the Coanda flap with the ducted fan and is a subject of future studies. The landing gear for the combustion configuration is allocated in pylons below the engine due to constraints of high wing aspect ratios, so the landing gear design is similar to the one considered by Heinze [47]. Although high uncertainties exist regarding mass penalties associated with both thrust vectoring and landing gear pylon weight, present research optimistically assumes that the weight penalty is small and is split in a way that both fuel cell and combustion configurations do not have a pronounced structural benefit because of one of the potential solutions. The leading edge for both configurations features a droop nose with sufficient sealing to allow efficient integration of the HLFC system and its performance for both sides of the wing.

All sizing tasks were performed for altitudes of 8000 and 12,000 m, as was discussed in Section 3. The fuel price for DOC estimation was selected equal to 4.86 EUR/kg as an average green hydrogen value obtained by Hoelzen [73].

Comparative Study Between Hydrogen Combustion and Fuel Cell Options

Tables 9–11 summarize geometric, mass, and propulsion system properties for all six airplanes sized during design space exploration.

Table 9. Comparison of medium-range hydrogen aircraft.

Parameter	LH ₂ Fuel Cell	LH ₂ Combustion	Units
Wing			
AR	11.0	13.0	-
b	43.0	42.0	m
S_w	168.3	135.3	m ²
Λ_{LE}	27.0	27.0	deg
C_r	7.78	6.43	m
C_t	2.33	2.12	m
Fuselage			
Length	37.5	37.5	m
Width	4.2	4.2	m
Height	5.06	4.88	m
Propulsion			
Number of engines	8	2	-
Engine diameter	1.37	1.94	m
Motor total power	2033		kW
Motor power-to-weight ratio	21.7		kW/kg
Sea-level static thrust	23.87	69.5	kN

Table 9. Cont.

Parameter	LH ₂ Fuel Cell	LH ₂ Combustion	Units
LH₂ tanks			
Total compartment length	20.12	20.12	m
Tank diameter	0.87	0.69	m
Weights			
m_{MTOW}	77.68	60.9	t
m_{empty}	53.48	37.7	t
$m_{engines}$	2.37	3.27	t
$m_{systems}$	15.92	14.3	t
$m_{LH2tanks}$	2.91	2.26	t
$m_{FCsystem}$	9.08		t
T/W	2.46	2.28	N/kg
W/S	461.5	450.2	kg/m ²

Table 10. Comparison of SE²A long-range hydrogen aircraft for B787 missions.

Parameter	LH ₂ Fuel Cell	LH ₂ Combustion	Units
Wing			
AR	10.5	12.5	-
b	71.9	65.8	m
S_w	492.2	346.0	m ²
Λ_{LE}	33.0	33.0	deg
C_r	14.38	11.7	m
C_t	2.87	2.93	m
Fuselage			
Length	56.8	56.8	m
Width	5.7	5.7	m
Height	8.23	7.75	m
Propulsion			
Number of engines	8	2	-
Engine diameter	1.59	2.81	m
Motor total power	8372		kW
Motor power-to-weight ratio	32.1		kW/kg
Sea-level static thrust	56.93	171.50	kN
LH₂ tanks			
Total compartment length	36.5	36.5	m
Tank diameter	2.38	1.90	m
Weights			
m_{MTOW}	228.7	162.9	t
m_{empty}	162.8	101.2	t
$m_{engines}$	5.45	7.93	t
$m_{systems}$	33.35	27.53	t
$m_{LH2tanks}$	16.98	12.83	t
$m_{FCsystem}$	37.34		t

Table 10. Cont.

Parameter	LH ₂ Fuel Cell	LH ₂ Combustion	Units
T/W	2.0	2.1	N/kg
W/S	464.6	470.8	kg/m ²

Table 11. Comparison of SE²A long-range hydrogen aircraft for B777 missions.

Parameter	LH ₂ Fuel Cell	LH ₂ Combustion	Units
Wing			
Wing AR	8.5	11.0	-
b	76.5	77.0	m
S_w	689.0	538.9	m ²
Λ_{LE}	33.0	33.0	deg
C_r	17.9	14.86	m
C_t	3.59	3.71	m
Fuselage			
Length	72.0	72.0	m
Width	6.6	6.6	m
Height	8.96	8.49	m
Propulsion			
Number of engines	10	2	-
Engine diameter	1.62	3.44	m
Motor total power	8899		kW
Motor power-to-weight ratio	32.6		kW/kg
Sea-level static thrust	63.87	279.7	kN
LH₂ tanks			
Total compartment length	48.6	48.6	m
Tank diameter	2.67	2.19	m
Weights			
m_{MTOW}	326.4	257.2	t
m_{empty}	221.9	159.2	t
$m_{engines}$	7.35	12.83	t
$m_{systems}$	50.45	42.97	t
$m_{LH2tanks}$	25.45	21.57	t
$m_{FCsystem}$	49.56		t
T/W	1.96	2.17	N/kg
W/S	473.7	477.3	kg/m ²

Several major trends among configurations can be observed. First, fuel cell systems were shown to add a significant amount of additional mass that promoted a snowball effect and eventually increased the MTOM and the fuel burn for all fuel cell-powered aircraft regardless of their size. Higher masses also influence the platform area and the aspect ratio that affect the drag and further influence the masses. The fuel cell system mass breakdown based on the fuel cell mass estimation method summarized in Equation (20) is summarized in Table 12 for each configuration. For each configuration, the mass of fuel cells contributes to almost 60% of the total system mass while the second heaviest component is the cooling system. A significant mass penalty then enforced a requirement of

a substantially larger wing planform area compared to hydrogen combustion configurations. Therefore, design aspect ratios for fuel cell configurations became 15% lower on average compared to combustion configurations. It can also be observed that mass penalties increase the power demand of aircraft proportional to their size. The heaviest aircraft requires almost 9 MW per motor with ten electric motors, which is a rather challenging requirement to achieve. It must also be noted that the present aircraft also features a range of future airframe technologies that significantly improve an airplane's energy efficiency. Propulsion system challenges will only be magnified if airframe technologies are not integrated or do not achieve desired performance levels. Aircraft thrust-to-weight ratios at the sea-level static thrust condition and wing loadings remain relatively similar regardless of the configuration. Both conditions were driven by either takeoff or maximum cruise speed requirements. Finally, the absence of span wingspan constraints substantially increased the values for medium- and long-range heavy fuel cell configurations. A wingspan increase in the order of 20% was observed compared to Annex 14 span regulations for each respective category. Such an increase in the wingspan may present challenges for future operations, and span reductions may significantly reduce the aspect ratio and may considerably diminish achieved fuel burn benefits. However, a lighter long-range fuel cell configuration presented a desired increase in wingspan of 10%, which could potentially be achieved using folding wingtips.

Table 12. Fuel cell system mass breakdown for sized configurations.

Component	Medium-Range	Long-Range B787-9 Category	Long-Range B777-300ER Category	Units
Fuel cells	5.3	21.7	28.8	t
Compressors	1.2	4.8	6.4	t
Cooling system	2.3	9.6	12.8	t
Humidifier	0.3	1.1	1.54	t
Total	9.0	37.3	49.6	t

A direct operating cost comparison is shown in Figure 12 and was based on average values between low- and high-altitude cases. Based on the figure, values of DOC show higher values for fuel cell aircraft, with an average value of 30%. Such a trend is motivated by a significantly higher airframe mass that is caused by the presence of heavy fuel cells and higher fuel costs.

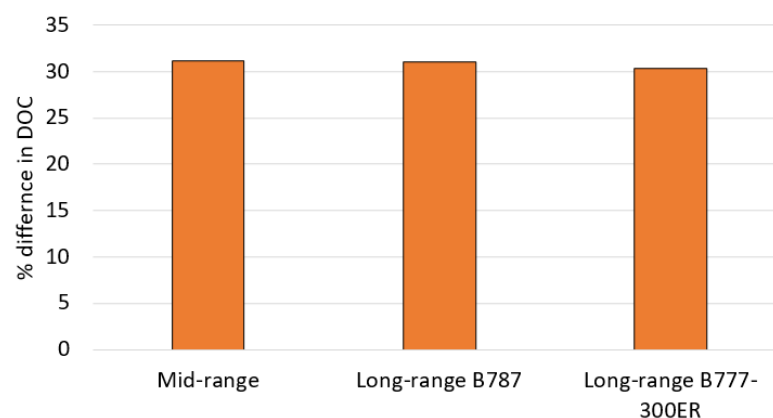


Figure 12. Comparison of average DOC between fuel cell and hydrogen combustion aircraft for three types of fleet.

Figures 13–15 compare equivalent emission results for fuel cell and combustion aircraft for three classes studied in the present work. In Figure 13, the in-flight breakdown of

emissions is shown for low and high-altitude cases. Since hydrogen aircraft do not emit CO_2 , only NO_x and contrail emissions are compared. Moreover, no NO_x emission is present for the fuel cell case. For medium-range aircraft, the contribution of NO_x is rather small compared to the contrails that generally dominate the emissions if advanced technologies and new energy sources are introduced. Since the average contrail formation depends on the altitude, higher-altitude cases produced more emissions on average. At low altitudes, the fuel cell case produced more contrails due to its higher total mass and higher fuel burn that affects the contrail formation, as presented in Figure 10. On the other hand, the difference between the combustion and the fuel cell cases is relatively small and equal to 6%, as shown in Figure 14. At higher latitudes, almost similar overall emissions for both aircraft were obtained.

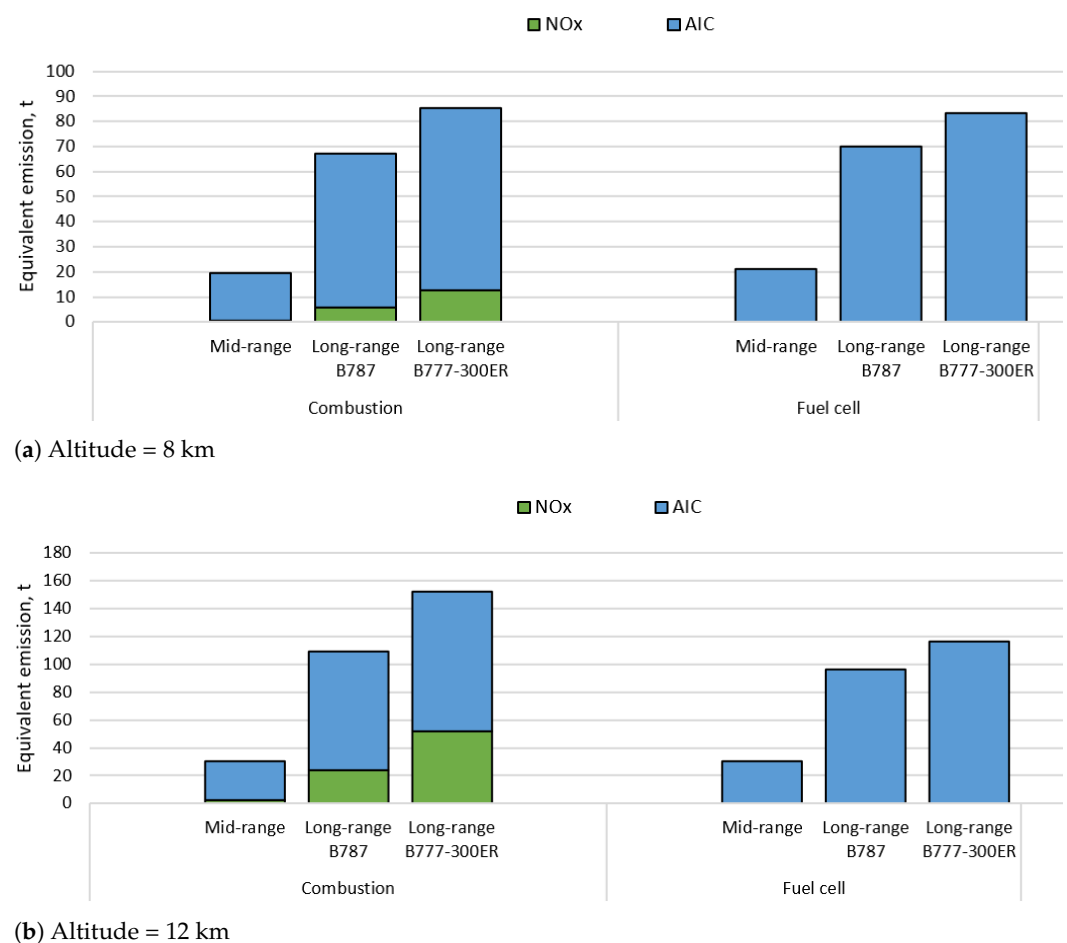


Figure 13. Comparison of emission components between the hydrogen combustion and fuel cell aircraft.

With the increase in airplane size and range, the contribution of NO_x becomes generally more pronounced, which negatively influences the overall emissions. Although the contrail emissions for the fuel cell configuration are larger than the combustion one due to substantially larger total mass, additional contributions of NO_x make the difference in overall emissions less pronounced and reach almost similar values, where the conclusion regarding the better design option becomes more challenging. Moreover, if less emissions favorable flight routes are used, such as an altitude of 12,000 m using the average emissions model, then the contribution of NO_x becomes so significant that the overall emissions of long-range fuel cell aircraft could be reduced by 14–24% compared to the hydrogen combustion option. Therefore, the fuel cell configuration for long-range applications may

be a highly valuable alternative to hydrogen combustion in cases when a fleet with robust operating capabilities at various altitudes from an emissions perspective is required.

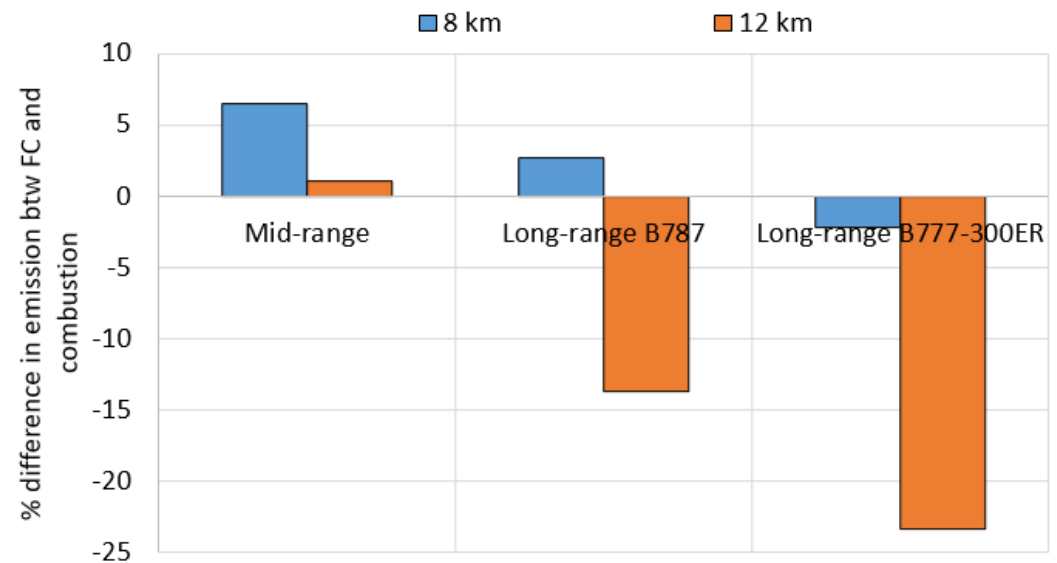


Figure 14. Comparison of in-flight equivalent CO₂ emissions between hydrogen combustion and fuel cell aircraft.

It must be pointed out again that the model used in the studies is the average one and does not represent local emission scenarios for particular latitude cases. For instance, medium-latitude cases are expected to have a lower emission level than the average one and should be closer to the one that represents an average lower-altitude case. However, an average performance is useful to represent a generic scenario for a large fleet with various route options.

Although emissions play an important role in the overall climate impact of the fleet, it is also important to understand how much fuel is required for a given aircraft and how much load on sustainable energy production is used to achieve minimum overall emissions. Figure 15 compares equivalent emissions and fuel burn for all hydrogen configurations studied in the present work and also compares them to the kerosene configuration studied in ref. [21]. Both scenarios for average low and high-altitude operations are considered. Although emissions for fuel cell aircraft could be similar for altitudes with lower impacts of NO_x emissions and could be substantially more environmentally friendly for altitudes with higher NO_x contributions, the overall energy demand is substantially larger for fuel cell aircraft. The difference between fuel cell and combustion aircraft fuel burn ranges between 22% and 30%, depending on the type of aircraft and the operational altitude. This could extend the availability of fully green hydrogen and may affect the overall emissions in the near term. If the hydrogen production starts deviating from the green one, the overall emission comparison between fuel cell and combustion configurations will shift more towards the hydrogen combustion scenario because it consumes less fuel and there will be fewer emissions during production. Therefore, the decision regarding the most sustainable aircraft configuration heavily depends on the hydrogen production strategy, given that aircraft technologies achieve the desired performance values.

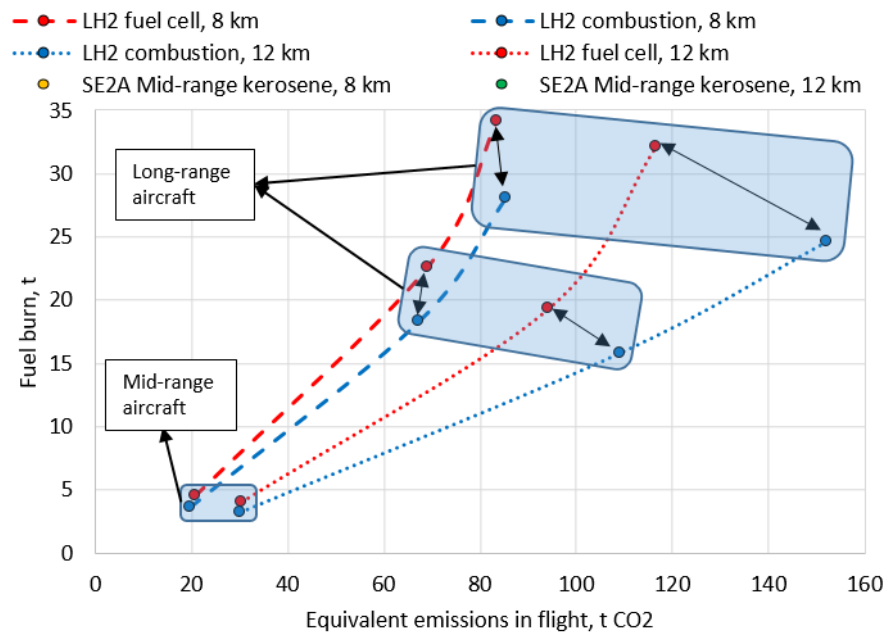


Figure 15. Comparison of in-flight fuel burn and emissions for three aircraft segments with fuel cells and hydrogen combustion configurations. The reference medium-range configuration corresponds to one described in ref. [21].

5. Conclusions

The present work investigated the potential of hydrogen fuel cell aircraft compared to the hydrogen combustion configuration in the context of a future fleet featuring advanced airframe and propulsion technologies. Three types of transonic aircraft were considered to investigate potential benefits and drawbacks for several market segments: medium-range, and two long-range aircraft of different sizes and ranges. A multi-fidelity initial sizing methodology based on the combination of constraint and mission analyses was created for fuel cell and combustion options to size each configuration and estimate equivalent CO₂ emissions and direct operating costs.

From the emissions perspective, operations at altitudes with low effects of NO_x make hydrogen combustion options potentially more favorable due to lower masses and, therefore, lower overall emissions compared to the fuel cell alternatives. On the other hand, flights at higher altitudes (that are more common for transcontinental flights and also show a more favorable region for lower contrail emissions) substantially increase NO_x emissions for hydrogen combustion configurations. Moreover, the difference in emissions between fuel cell and combustion configurations for similar configurations at similar flight conditions rapidly increases with aircraft size and can reach values of up to 24% for large long-range aircraft. This fact indicates that fuel cell aircraft could be potentially more robust from the operational perspective due to the lack of NO_x emissions, which are highly sensitive to operating altitudes. Therefore, hydrogen fuel cell configurations for long-range applications may be a more favorable solution compared to hydrogen combustion configurations.

Although favorable results were achieved for fuel-cell configurations, it must be noted that energy and propulsion system requirements remain highly challenging for fuel-cell-powered aircraft. Particularly, long-range aircraft require high-power motors of the order of 8–9 MW per engine. A general question of the possibility of designing a compact motor of such power is present, and additional research is required to answer this question. Moreover, challenges related to systems integration and corresponding aircraft design problems need to be studied further to ensure that a more comprehensive understanding of the problem is achieved. Particularly, the proper modeling and design of thermal management

systems are of high importance to further understand their size and mass. If significantly higher mass and drag of the system is obtained, then emission and cost gains will increase proportionally, making any potential applications even less feasible. It also needs to be noted that the present work also considered potential future technologies that artificially made propulsion system requirements less challenging. Moreover, energy system assumptions were also based on potential performance values that could be achieved by 2050. Achieving the given performance of airframe and propulsion technologies also remains challenging, and it is necessary to ensure that fuel cell configurations can be introduced as early as possible. The present work assumed rather optimistic values to explore more optimistic scenarios. To ensure that a more realistic assessment can be made, more precise modeling of each technology is required. The problem of operating costs needs to be addressed as well, which was shown to be in the order of 30% higher compared to hydrogen combustion configurations for all fuel cell alternatives. One potential solution to reduce costs could be to introduce fuel cell technologies that could substantially reduce the fuel cell system mass and the DOC. It must also be noted that in-flight emissions do not represent the overall life cycle emissions of the aircraft and the selection of the aircraft configuration highly depends on the way hydrogen is produced. The benefits of fuel cell aircraft can be maximized with green hydrogen production, while deviations from less sustainable fuel production will move the design selection closer to the hydrogen combustion variant. Therefore, green hydrogen production is required to achieve favorable emissions for fuel cell aircrafts. Future research should focus on the potential and various options of economically feasible green hydrogen production as one of the key drivers of the fuel cell configuration for commercial applications. Finally, emission results highly depend on the modeling approach which remains uncertain. The model uncertainty related to contrail emissions shall be mitigated in future work for hydrogen combustion and fuel cell propulsion systems.

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Abbreviations

The following abbreviations are used in this manuscript:

Λ_{LE}	Leading edge sweep in deg
Re	Reynolds number
X_{tr}	Transition chord ratio
m	Mass in kg

W/S	Wing loading in kg/m ²
S	Area in m ²
D	Diameter in m
AR	Aspect ratio
R	Radius in m, Resistance in Ohm
r	Area-specific resistance in Ohm/m ²
s	Emission forcing factor
F	Faraday constant
H	Height in m
W	Width in m
C	Chord in m
T/W	Thrust-to-weight ratio in N/kg
C_p	Specific heat at constant pressure in J/kg/K
b	Span in m
λ	Taper ratio
V	Volume in m ³ , Velocity in m/s
V_c	Fuel cell voltage in V
ρ	Power density in kW/kg
P	Power in kW, Pressure in Pa
HP	Horsepower
A	Cross-sectional area in m ²
T	Temperature in K, Thrust
$\dot{Q}$	Heat transfer rate in J/s
σ	Stress in Pa
e_w	Weld efficiency
DOC	Direct operating costs in EUR
p	Price in EUR
$\dot{m}_f$	Fuel flow rate in kg/s
t	Thickness in m
η	Efficiency
i	Current density in A/m ²
E_{oc}	Open circuit voltage in V
AIC	Aviation-induced cloudiness
FC	Fuel cell
$HLFC$	Hybrid laminar flow control
NLF	Natural laminar flow
SAF	Sustainable aviation fuel
PEM	Proton exchange membrane
$PCHP$	Phase-change-heat-pump
BOP	Balance of plant
$CFRP$	Carbon fiber-reinforced plastic
ISA	International Standard Atmosphere
$UHBPR$	Ultra-high bypass ratio
SFC	Specific fuel consumption
TMS	Thermal management system
$PMAD$	Power management and distribution
GH_2	Gaseous hydrogen
LH_2	Liquid hydrogen
$MTOM$	Maximum takeoff mass
MLM	Maximum landing mass
CG	Center of gravity

Appendix A. Derivation of the Hydrogen Fuel Cell Energy Network for Aircraft Sizing

Derivation of Figure A1 shows a schematic fuel cell diagram with all major components.

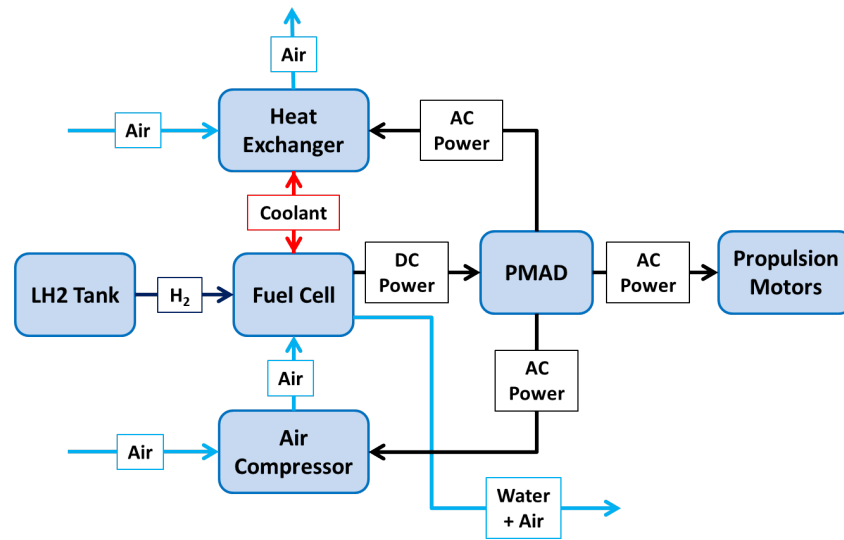


Figure A1. Schematic representation of a fuel cell energy network.

The power of the fuel cell, including its major components, can be defined by

$$P_{FC} = P_{el_{net}} + P_{comp} + P_{cool} = \frac{P_p}{\eta_p \eta_{GB} \eta_{PMAD} \eta_{EM}} + P_{comp} + P_{cool} \quad (A1)$$

where P_{FC} is the fuel cell power, P_{shaft} is the shaft power, P_{comp} is the compressor power, P_{cool} is the power of the cooling system, and $P_{el_{net}}$ is the electric network power. The power generated by the propeller P_p is corrected by the propulsive efficiency η_p , the gearbox efficiency η_{GB} , the electric motor efficiency η_{EM} , and the power management efficiency η_{PMAD} .

The fuel cell power at the cell level can be represented as

$$P_{FC} = V_c \cdot i \cdot A_{FC} \quad (A2)$$

where V_c is the fuel cell voltage, i is the current density, and A_{FC} is the fuel cell active area. The fuel cell voltage depends on the current density and can be generally defined using various relations. Generally, the fuel cell voltage is defined by

$$V_c = E_{oc} - \eta_{act} - \eta_{ohm} - \eta_{conc} \quad (A3)$$

where η_{act} are activation losses, η_{ohm} are ohmic losses, and η_{conc} are concentration losses. Based on derivations described in [52], the fuel cell voltage can be simplistically defined by

$$V_c = E_{oc} - ir - \frac{RT}{2\alpha F} \ln(i) + me^{ni} \quad (A4)$$

where E_{oc} is a reversible open circuit voltage (or ideal voltage), r is the area-specific resistance, i is the current density, and m and n are correction factors that describe the mass-transfer overvoltage. A fraction before the first logarithm is the Tafel slope and consists of the gas constant R , operating temperature T , the Faraday constant F , and the transfer coefficient at the cathode α set to 0.3. Equation (A4) describes the voltage as a function of the current density.

Finally, the fuel cell's electric efficiency is defined according to ref. [52] by

$$\eta_{FC} = \mu_f \frac{V_c}{E_{oc}} \quad (A5)$$

where μ_f is the fuel utilization coefficient, usually taken as 0.95. Knowing the properties of a fuel cell, Equation (A4) is solved for the fuel cell voltage for a range of current densities. Then, the fuel cell power per active fuel cell area is computed using Equation (A2). From the obtained values of power per cell area, the maximum value is extracted and the database of ratios of fuel cell power to its maximum value is created. Fuel cell efficiency is also computed using Equation (A5). All calculated data are combined into one plot, where the voltage can be described as a function of the power ratio with respect to the maximum available power, as shown in Figure A2.

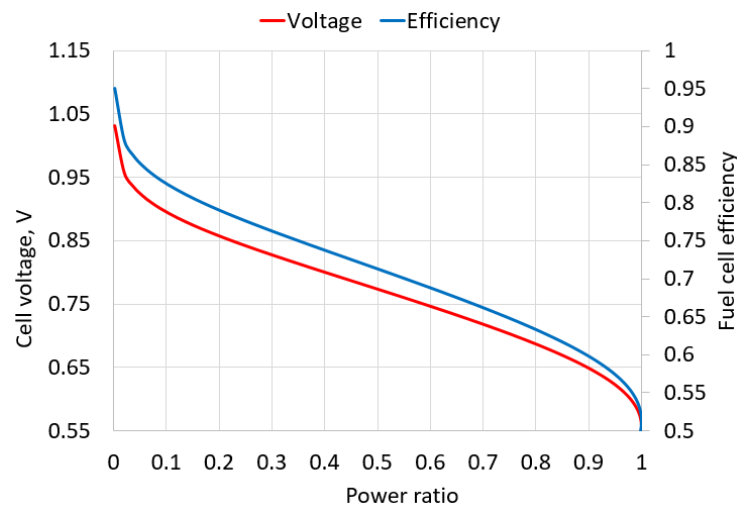


Figure A2. Fuel cell efficiency and voltage as a function of the power ratio for a sample fuel cell.

Assuming that the fuel cell shall operate at current densities that do not exceed the maximum power ratio, the overall fuel cell thermal efficiency can be directly described as a function of the power ratio. Fuel cell efficiency will be used for other formulations described below.

Returning to Equation (A2), the shaft power is provided by the power required during the flight at a given altitude and airspeed, which depends on the type of the propulsor and is defined by

$$P_{shaft} = \frac{TV}{\eta_p} \quad (A6)$$

where T is the thrust at a given time period, V is the flight airspeed, and η_p is the propulsive efficiency that can be defined using either the cubic spline method or the momentum theory at early design stages [58]. Values of η_{PMAD} and η_{EM} depend on the selection of the PMAD and the motor, respectively.

The compressor power required for fuel cell operations at appropriate pressure levels is defined from ref. [52] by

$$P_{comp} = \frac{\dot{m}_{air} C_{p_{air}} T_1}{\eta_c \eta_{EM}} \left(\left(1.05 \frac{P_2}{P_1} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right) \quad (A7)$$

where $C_{p_{air}}$ is the heat capacity of the air, $\dot{m}_{air}$ is the air mass flow rate through the compressor, T_1 is the ambient temperature, η_c is the compressor efficiency, and P_1 and P_2 are pressures before and after the compressor, respectively. The mass flow rate is defined by

$$\dot{m}_{air} = 2.856 \cdot 10^{-7} \cdot \lambda_{O_2} \cdot \frac{P_{FC}}{V_c} \quad (A8)$$

where λ_{O_2} is the ratio of oxygen supplied versus the oxygen necessary for the electrochemical reaction with hydrogen, which ranges between 1.5 and 2.0.

The cooling system power depends on the operating temperature of the fuel cell and the amount of heat to be rejected. The heat power rejected by the fuel cell is defined according to ref. [52] by

$$P_{heat,reject} = \left(\frac{1}{\eta_{FC}} - 1 \right) P_{FC} \quad (A9)$$

The power required for the cooling system is based on the work of NASA [50] and is defined by

$$P_{cool} = \left(0.371 P_{heat,reject} + 1.33 \right) \cdot f(dT) \quad (A10)$$

$$f(dT) = 0.0038 \left(\frac{T_{air}}{dT} \right)^2 + 0.0352 \left(\frac{T_{air}}{dT} \right) + 0.1817 \quad (A11)$$

where $P_{heat,reject}$ is defined in kW.

All elements involved in Equation (A1) depend on the fuel cell power, its efficiency, or the fuel cell/power ratio. Therefore, an iterative solution is required to converge the required fuel cell power at a given time segment along the mission.

Equation (A1) is used for both the fuel cell operation simulation along the flight and for the overall system sizing. For a given required maximum propeller or ducted fan power, the fuel cell sizing is performed using the idea of establishing the most demanding operating conditions and then sizing each fuel cell system component according to those conditions. The maximum fuel cell power is then defined by,

$$P_{FC_{max}} = k_{oversize} (\max(P_{el_{net}}) + \max(P_{comp}) + \max(P_{cool})) \quad (A12)$$

where $k_{oversize}$ is the fuel cell oversize factor that may make the fuel cell operate at higher efficiencies. The approach shown in Equation (A12) accounts for the maximum power of each component for maximum at worst operating conditions. Worst operating conditions of sea level and service ceiling were prescribed for the cooling system and compressors, respectively.

Finally, the fuel burn at the fuel cell is defined using the relation provided in ref. [12],

$$\dot{m}_f = \frac{P_{FC}}{\eta_{ideal} \eta_{FC} C_{H_2}} \quad (A13)$$

where C_{H_2} is the liquid hydrogen power density, η_{ideal} is obtained from ref. [52] and equals 0.83, and P_{FC} and η_{FC} are obtained using Equations (A2) and (A5), respectively.

The method can be applied for the powertrain or overall aircraft sizing (using the constraint analysis) and mission simulation. Note that the fuel cell efficiency curve used for the airplane sizing can be obtained using Equation (A4) with given constants or from the external database. The present work currently focuses on the formulation of Equation (A4).

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