

CRANFIELD UNIVERSITY

Pavlos A. Rompokos

Turboelectric Distributed Propulsion System Design and Tank Sizing
For Novel Hydrogen-Fuelled Aircraft: A Mission-Level Analysis

School Of Aerospace, Transport And Manufacturing
PhD in Aerospace

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Academic Year: 2019 - 2022

Supervisors: Dr Devaiah Nalianda, Dr Andrew Rolt
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Abstract

In an ever-growing civil aviation industry, the need to reduce the related carbon footprint is more relevant than ever. One way to substantially contribute to this cause is to use liquid hydrogen (LH₂) as a fuel, which would allow for zero in-flight CO₂ emissions. However, there are major challenges associated with its implementation, both from the perspective of the airframe design as well as the propulsion and fuel system architecture.

This research contributes to knowledge by addressing the sizing of the LH₂ aircraft components from a mission level perspective. More specifically two novel aspects are addressed based on that philosophy: LH₂ tank sizing with gaseous hydrogen (GH₂) extraction for pressure control and design space exploration of turboelectric distributed propulsion (TeDP) and partially TeDP configurations powered by LH₂.

This is achieved through a developed framework that enables the assessment of a hydrogen aircraft and its subsystems at mission level. It comprises four co-dependent modules: aircraft performance, aircraft weight estimation, LH₂ tank sizing and propulsion system performance and weight estimation. For each one of the modules, novel and existing methodologies, from kerosene applications, are combined to address the sizing of the components of the LH₂ aircraft.

The research then focuses on a case study of a blended-wing body architecture that is based on NASA's N3-X and is powered by a TeDP system. The airframe is scaled to accommodate more passengers as well as allow for enough volume to fit large enough LH₂ tanks that would enable the aircraft to meet design mission requirements.

Based on the available airframe volume, two sizing philosophies are considered: one where only LH₂ is used for the engines and another where both LH₂ and GH₂ are used to restrain the pressure increase of the tank during the cruise segment. Compared to the first application, the second showcased both significant weight savings as well as increased fuel capacity for the same external volume occupied.

Finally, for the propulsion system, a design space exploration is performed to identify optimum propulsion design parameters for turboelectric and partially turboelectric distributed propulsion configurations. The different design options are analysed based on efficiency, weight and mission level fuel economy on the design mission and it is demonstrated that higher specific thrust designs are more suitable for the LH₂-powered long-range application presented. The

optimum configuration identified consists of two low by-pass ratio turbofans that are buried in the wing roots and power 12 fans installed over the aft section of the fuselage.

Keywords:

LH₂, Hydrogen Propulsion, Tank Sizing, Hydrogen Venting, Turbo-electric Distributed Propulsion (TeDP), Partially TeDP, Multipoint Propulsion System Sizing, Variable Area Nozzle, Propulsion System Weight Estimation, Aircraft Weight Estimation, Blended-Wing Body, Boundary Layer Ingestion

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List of Abbreviations

AC	Alternating Current
AP	Aircraft Performance
APU	Auxiliary Power Unit
ASME	American Society of Mechanical Engineers
ATI	Aerospace Technology Institute
AWE	Aircraft Weight Estimation
BL	Boundary Layer
BLI	Boundary Layer Ingestion
BPR	By-pass Ratio
BWB	Blended Wing Body
CC	Combustion Chamber
DC	Direct Current
DP	Design Point
EIS	Entry Into Service
ENABLEH2	ENABLING cryogENic Hydrogen based CO2 free air transport
FLOPS	Flight Optimization System
FoS	Factor of Safety
FPR	Fan Pressure Ratio
GH2	Gaseous Hydrogen
HP	High Pressure Spool
HTS	High Temperature Superconducting
HX	Heat Exchanger
IP	Intermediate Pressure Spool
LH2	Liquid Hydrogen
LNG	Liquefied Natural Gas
LPT	Low Pressure Turbine
LR	Long Range
LTS	LH2 Tank Sizing
MAG	Mass Flow Group
MTOW	Maximum Take-off Weight
NIST	National Institute of Standards and Technology
NPF	Net Propulsive Force
OD	Off-design
OWE	Operating Weight Empty
PSS	Propulsion System Sizing
SAF	Synthetic Aviation Fuel
SAF	Sustainable Aviation Fuel

SLS	Sea level Static
SMR	Short to Medium Range
syn-Jet	Synthetic Jet Fuel
T&W	Tube and Wing
TeDP	Turboelectric Distributed Propulsion
TF	Turbofan
TOW	Take-off Weight
TS	Turboshaft
UAV	Unmanned Aerial Vehicle
VAN	Variable Nozzle Area

Nomenclature

Symbol	Units	Description
A	m ²	Area
B _g	T	Peak of fundamental magnetic flux waveform at the airgap
B _r	T	Remnant flux density of the magnet
C ₀₋₃	-	Power loss constants
C _f	-	Skin friction coefficient
C _{min}	W/K	Heat capacity rate
C _p	J/kg/K	Specific heat capacity
D	m	Diameter
d	m	Depth
D ₂	m	External diameter of the tank
E	Pa	Modulus of elasticity
EF	-	Heat exchanger effectiveness
e _w	-	Weld efficiency
f	Hz	Current frequency
FD	N	Drag force
FG	N	Gross thrust
FN	N	Net thrust
g	m/s ²	Gravitational acceleration
h	W/m ² /K	Convective heat transfer coefficient
h	m	Height
h _{lg}	J/kg	Specific enthalpy change of vaporisation
HTR	-	Hub to tip ratio
I	kg m ²	Moment of Inertia
J	A/m ²	Current density of the conductor
k	W/m ² /K	Thermal conductivity
k _c	-	Conductor packing factor
k _l	-	Winding factor
K _s		Electrical loading
k _s	-	Safety factor
l	m	Length
M	kg	Mass
m	kg/s	Mass flow
N	rad/s	Rotational speed

Nu	-	Nusselt number
P	Pa	Pressure
p	-	Number of pole pairs
Pr	-	Prandtl Number
PV _{GB}	kg	Gearbox parametric value for weight estimation
PW	W	Power
Q	W	Heat transfer rate
Q _R	Nm	Rotor torque
Q _w	W	Heat addition rate
Ra	-	Rayleigh number
Re	-	Reynolds number
R _{th}	K/W	Thermal resistance of the insulation
s	J/kg/K	Specific entropy
S	m ²	Surface area
SEC	N/W	Specific energy consumption
SFC	N/kg/s	Specific fuel consumption
SFN	m/s	Specific thrust
T	K	Temperature
t	m	Thickness
T _f	K	Fuel temperature
t _{in}	m	Insulation thickness
T _{s1}	K	Tank internal surface temperature
T _{s2}	K	Tank external surface temperature
t _w	m	Wall thickness
u	J/kg	Specific internal energy
V	m ³	Volume
ν	m ² /s	Kinematic viscosity
V	m/s	Velocity
w	m	Width
W	kg/s	Mass flow rate
W _{cor}	kg/s	Mass flow rate corrected for temperature and pressure
x	-	Mixture Quality
α	rad	Angle
γ	-	Heat capacity ratio
δ	m	Boundary layer thickness
ε	-	Emissivity

η	-	efficiency
ρ	kg/m ³	Density
σ	W/m ² /K ⁴	Stefan boltzmann constant
σ_R	Pa	Shear stress of at the surface of the rotor
σ_y	Pa	Yield strength
σ_α	Pa	Maximum allowable stress
Φ	Pa/J/m ³	Energy derivative
ω	rad/s	Angular speed

1 Context and research scope

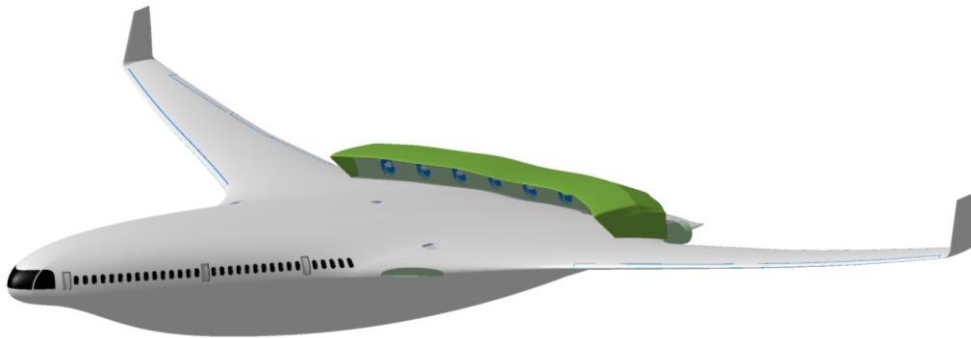


Figure 1-1 EH2 BWB concept, an LH₂-powered long range commercial aircraft, sized in the context of this work

1.1 Introduction and motivation

Throughout the various industries associated with CO₂ emissions, aviation currently accounts for a relatively small part but is predicted to proportionally increase over the coming years [1]. This may occur as the power generation and automotive industries, for example, are decarbonising faster than the aviation industry, which still heavily depends on fossil fuels. Reducing CO₂ emissions is mainly achieved by improvements in propulsion system and airframe architecture but considering the potential growth in aviation, these alone are insufficient. An alternative way is the use of hydrogen which upon combustion or when used in fuel cells allows for zero CO₂ emissions.

In recognition of the hydrogen potential, Airbus announced the Zero-e project [2] in early 2021 and presented three short-range concepts that would be the first commercial aircraft that would use liquid hydrogen as a fuel (LH₂). They are expected to enter in service by 2035. The first concept has a tube and wing (T&W) configuration, features two “hybrid” turboprop engines and is capable of carrying 100 passengers for at least 1000 nmi. The other two could carry up to

200 passengers for 2000 nmi. One concept features a T&W, the other a blended-wing body (BWB) architecture and both are powered by hybrid turbofans. To mature the technologies required to enable such LH₂ powered flights, Airbus in collaboration with CFM, at the time of writing, is also planning to create a demonstrator and test LH₂ storage, distribution and combustion on a conventional gas turbine. Among many others over this recent period, this is a significant step towards commercial hydrogen flight but certainly not the first.

LH₂ as an aviation fuel has been considered as early as 1923 when Meyer produced a report [3] identifying the LH₂ heat release benefits as well as the volumetric disadvantages compared with gasoline. Apart from the heat release, Jonash et al. [4] in 1955, through experimental work, demonstrated that gaseous hydrogen has wide flammability limits at low pressure, allowing for stable combustion in a jet engine at very high altitude.

In acknowledgement of these properties, the NACA in 1955 conducted preliminary designs of bombers and reconnaissance aircraft for very high altitude, both for subsonic and supersonic applications [5]. Immediately afterwards, Silverstein led the Bee project [6] where a Martin B-57 Canberra was converted to operate with gaseous hydrogen on one engine. External tanks were mounted under the wingtips and the LH₂ fuel was preheated in a heat exchanger before reaching the engine. Three test flights were completed with the B-57B taking off with conventional jet fuel, switching to hydrogen at cruise and switching back to kerosene for landing. The hydrogen work of that period is covered in detail by Sloop [7].

The latest prominent demonstrations of hydrogen-powered flight came from Boeing and ZeroAvia. Boeing introduced a long-endurance UAV, the Phantom Eye, featuring two large spherical LH₂ tanks installed in the fuselage [8, 9]. Between 2012 and 2014 nine test flights were performed and the aircraft cruised at 54000 ft for 8 hours. Despite the successful demonstration, the project was discontinued and it is unclear whether there will be any long-endurance successor UAVs. ZeroAvia, founded in 2018, converted a six-seater aircraft to run on hydrogen via fuel cells and in 2020 performed the “world

largest hydrogen-electric flight” to date [10]. The company currently is working to scale up the application and convert a Dornier 228 19-seater regional aircraft with the same powertrain.

Considering the ever-growing interest in LH₂ for aircraft propulsion, the need to have tools and methods to estimate the performance of such aircraft and its subsystems is of primary significance. The main driver of this project is to contribute to that purpose at the preliminary stage of the design process.

1.2 Aim and objectives

The aim of this work is to develop a multidisciplinary framework for the assessment of LH₂-fuelled aircraft and their subsystems. To achieve that the following objectives are set:

1. Use aircraft performance and weight estimation tools to size and perform mission analysis on a selected LH₂ aircraft platform.
2. Develop an LH₂ tank sizing tool to assess the LH₂ tank performance throughout the flight envelope and introduce a methodology to optimally size any tank design based on airframe integration and design mission requirements.
3. Create a multidisciplinary platform for turbo-electric distributed propulsion system performance and weight estimation.
4. Develop a framework and integrate the tools developed for mission level optimisation of the LH₂ aircraft.

1.3 Novelties introduced and contribution to knowledge

This work has made significant contributions to the preliminary sizing of the subsystems of an LH₂-powered aircraft considering mission level performance. Through the developed framework the co-dependent sizing parameters between aircraft, propulsion and the fuel system are identified and two novelties are introduced. The first is the assessment of turboelectric and partially turboelectric distributed propulsion (TeDP) configurations based on mission

level performance of an LH₂-powered aircraft. The second is a methodology for LH₂ tank sizing and optimisation with different pressure control mechanisms considering the design mission requirements.

According to the literature survey conducted, previous studies on conceptual LH₂ aircraft have addressed the design of turbofans based on engine efficiency. At the same time, TeDP assessments in the past, although mostly focused on the performance of the distributed propulsors, have demonstrated that a combination of propulsion system efficiency and weight would allow for maximum mission fuel economy. In this study, the existing principles for kerosene-based propulsion system sizing are adapted to address for the first time in reported literature the design of LH₂-powered TeDP and partially TeDP configurations with synchronous electrical transmission and variable nozzle area technologies. The key outcome is that higher specific thrust designs although less efficient than lower specific thrust ones, allow for greater mission fuel economy on LH₂-powered aircraft due to lower weight.

For the LH₂ tanks, existing literature has mostly addressed the effect of the tank geometry and material selection on the tank sizing. In this context, a new dimension to the design process is introduced that allows for optimum tank designs whether GH₂ extraction for pressure relief is used or not and the differences between the two of them are quantified. In addition, the unusable LH₂ that is evaporated throughout a given mission and remains in the tank at the end of it and its effect on the tank design is also addressed. Through the examined application, it was demonstrated that for determined external volume, hydrogen tanks that utilise systematic GH₂ extraction for pressure relief are potentially lighter and can store more LH₂ than tanks without that feature.

1.4 Thesis structure

The thesis herein is divided into 5 chapters:

Chapter 2: The literature review of this research is presented. This chapter is subdivided into four sections that analyse previous work on key areas of focus

for this study. These are previous studies of conceptual LH₂ aircraft, BWB airframe architectures, turboelectric distributed propulsion and LH₂ tanks for aircraft applications. At the end of every section, the relevant research questions identified are also presented.

Chapter 3: Includes a detailed description of the mathematical models and methodologies used to assess the hydrogen aircraft. It is subdivided into four sections that address each objective set, namely, integration framework, aircraft sizing and mission performance, LH₂ tank sizing and propulsion system sizing.

Chapter 4: Presents the sizing results of an LH₂-fuelled BWB with TeDP. It includes seven subsections that discuss, the aircraft requirements set, the sized airframe geometry, the LH₂ tank results, the propulsion system sizing inputs and the design space exploration of two TeDP configurations. The final section discusses the performance of the aircraft based on the optimum propulsion system design selected from the design space exploration.

Chapter 5: Summarises the conclusions drawn from this study and lists recommendations for future work.

Apart from the main chapters, a list of references is included after chapter 5 as well as two Appendices with subsidiary material related to this work.

1.5 Publications

The author produced two papers that were presented at ASME Turbo EXPO conferences in 2020 [11] and 2021 [12], and one of these was subsequently published in the Journal of Engineering for Gas Turbines and Power [13]. The content for both of these papers was developed as part of the author's involvement in the ENABLEH2 project and this work. A further paper published in the Energies journal [14] used some of the tools and methodologies developed and discussed herein but in a different context. The publications are:

1. Rompokos P., Rolt A., Nalianda D., Sibilli T., Benson C., Cryogenic Fuel Storage Modelling and Optimisation for Aircraft Applications, ASME Turbo Expo 2021: Turbomachinery Technical Conference and Exposition (June 2021).
2. Rompokos P., Rolt A., Nalianda D., Isikveren A.T., Senné C., Grönstedt T., Abedi H., Synergistic Technology Combinations for Future Commercial Aircraft Using Liquid Hydrogen, Journal of Engineering for Gas Turbines and Power 143(7) (January 2021).
3. Rompokos P., Kissoon S., Roumeliotis I, Nalianda D., Nikolaidis T., Rolt A., Liquefied Natural Gas for Civil Aviation, Energies 13 (22) (November 2020).
4. Rompokos P., Rolt A., Nalianda D., Isikveren A.T., Senné C., Grönstedt T., Abedi H., Synergistic Technology Combinations for Future Commercial Aircraft Using Liquid Hydrogen, ASME Turbo Expo 2020: Turbomachinery Technical Conference and Exposition (September 2020).

2 Literature Survey

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In this chapter the literature review is presented and is divided into four subsections: previous work on conceptual LH₂ aircraft, a summary of key studies related to the blended wing body concept airframe, an overview of turbo-electric distributed propulsion and its subcomponents, and lastly a review of LH₂ tanks for aircraft applications. Although substantially more could be reported about the topics addressed, only a summary of the relevant material and the research questions are presented.

2.1 Conceptual studies of LH₂ aircraft

Brewer's studies

During the Middle East oil embargo in the 1970s, NASA funded more studies [15, 16] at Lockheed to investigate the use of alternative fuels for aviation, such as hydrogen, methane or synthetic jet (syn-Jet) fuel from coal. The studies led by Brewer and summarised in [17], involved the conceptual design of long-range subsonic and supersonic commercial aircraft with these three fuels. The first, and one of the most significant issues tackled in that project, was the LH₂ tank position. As LH₂ occupies significantly more space than kerosene, the aircraft designer needs to take into account the tank integration and positioning, as different design options may lead to weight and efficiency penalties. Brewer tackled this at the first stages of the study by conducting a qualitative analysis of 8 down-selected tube and wing (T&W) designs, of which he rejected 6 involving:

1. A design where the tanks were mounted on top of the passenger fuselage, due to maximum passenger exposure to fuel tanks
2. Two designs with a single tank integrated inside the fuselage and aft of the pressurised passenger cabin, due to excessive trim drag associated with the centre of gravity travel
3. A design with external podded tanks supported by a wing and the horizontal stabiliser
4. A twin fuselage configuration where the tanks were integrated into the mid-wing section, due to resulting large wetted area and high structural weight
5. A design with extended wing planform to integrate the tanks inside the wing, due to excessive weight and limitations on high-speed cruise

Two were selected for further development and are presented in Figure 2-1, The first featured two tanks inside the fuselage, one in front and one aft of the passenger cabin, schematic (a) of Figure 2-1. For the second the tanks were mounted in pods above the wings, schematic (b) of Figure 2-1. Following their preliminary design, the configuration with the internal tanks proved to be 16.4% more efficient due to lower weight and improved aerodynamic performance for

the same design mission. This configuration was then used for the detailed sizing of the whole aircraft including the fuel containment and distribution system [18, 19].

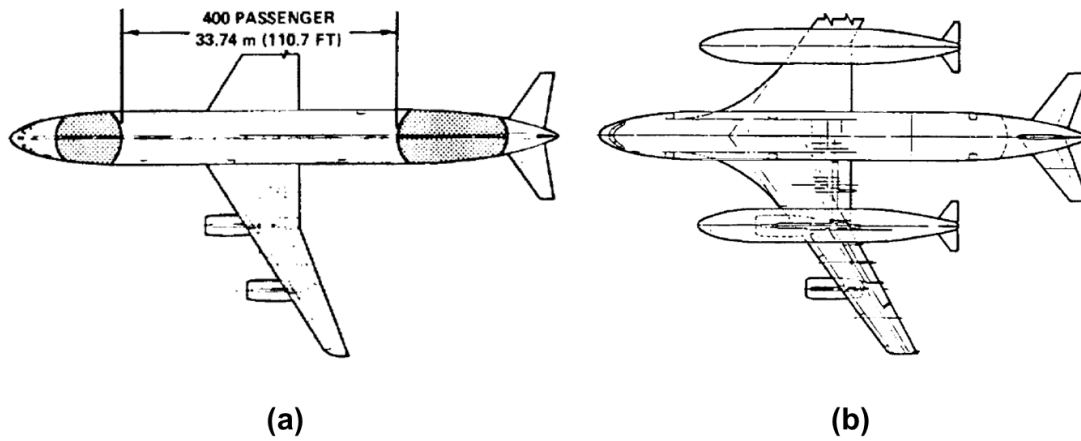


Figure 2-1 Down-selected T&W LH₂ aircraft adopted from Brewer [15]

Although the initial sizing requirements selected were for 400 passengers and 5500 nmi, the studies were extended and a few more aircraft were sized for 130 – 400 passengers and a range between 1500 and 10000 nmi for the three alternative fuels. For all the applications the energy reduction between the LH₂ and the syn-Jet configurations, as reported in [20], are presented in Table 2-1. Notably, as the design range and payload increase, higher energy savings compared to the syn-Jet applications can be obtained. This establishes the long-range application even more attractive to utilise LH₂. It is also worth mentioning that the 400-passenger for 10000 nmi case was considered to allow for considering a round trip of half of the range without refuelling, a design choice that heavily penalises the kerosene aircraft.

Table 2-1 Design mission energy benefits of LH₂ use compared with syn-Jet, obtained from [20]

Passengers	130	200	400	400	400
Range	1500	3000	3000	5500	10000
LH ₂ to syn-Jet Energy reduction	-2.5%	-5.0%	-7.0%	-16.4%	-32.9%

The key conclusions of interest derived from the Brewer studies are that the LH₂ aircraft are generally expected to:

1. have a heavier operating empty weight (OWE) due to the added weight of the LH₂ tanks
2. require less mass of fuel compared with kerosene, due to the different properties and as a result lower maximum take-off weight (MTOW)
3. have a penalised aerodynamic performance due to the added volumes required by the large cryogenic tanks

However, it is noted that the above-listed observations are generic guidelines and apply only for the assumptions and design choices considered by Brewer.

Cryoplane

In the late 1990s, Airbus led a project called Cryoplane [21]. Aircraft concepts covering all categories of commercial aviation were assessed and are summarised by Westenberger in [22]. Other issues such as airport and environmental compatibility, safety and medium/long term scenarios for a smooth transition from kerosene to hydrogen in aviation were also addressed.

Contrary to the Brewer studies, the Cryoplane considered that for the regional and short-range applications an LH₂ tank aft of the passenger cabin would not introduce significant problems related to centre of gravity shift throughout the flight. Some of the configurations are presented in Figure 2-2. The schematics shown were created by the author based on the Cryoplane configurations [21, 22]. The small regional aircraft, shown in schematic (a) of Figure 2-2, featured a single tank at the aft of the fuselage. The short/medium-range aircraft featured tanks above the fuselage in addition to the tank at the back, and is presented in schematic (b) of Figure 2-2. The long-range application had the same tank positioning as the equivalent one from Brewer but featured only two turbofans, picture (c) of Figure 2-2.

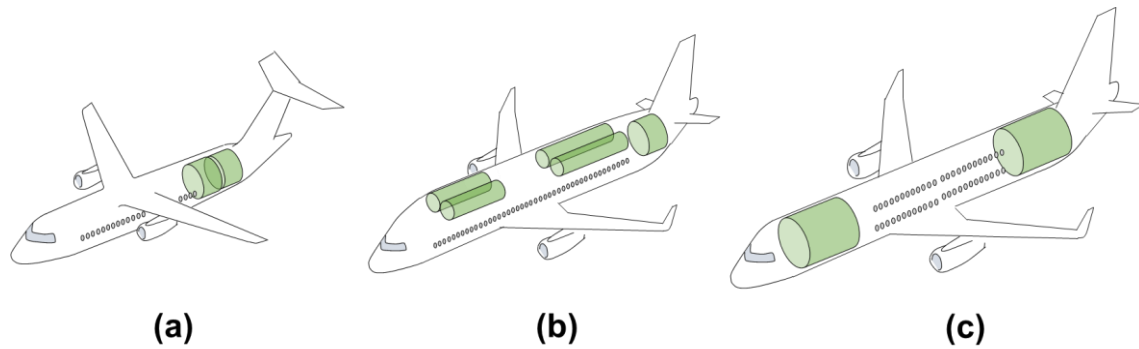


Figure 2-2 (a) small regional, (b) short-medium range and (c) long-range LH₂ concepts proposed by Cryoplane [21]

When comparing the LH₂ concepts with equivalent kerosene versions, the Cryoplanes also demonstrated an increase in OWE and a slight decrease in MTOW (apart from one configuration). However, the energy consumed for the design mission was found to be higher than that with kerosene, but the penalty seemed to be decreasing for the larger and longer-range aircraft. An exception to that rule is their reported “very large-long range” application but due to limited information available related to the LH₂ aircraft sizing process no conclusions are drawn by the author. The Cryoplane report recommended that further improvement in the long-range application can be provided by adopting a blended wing body (BWB) airframe. This concept was introduced a few years earlier by Liebeck et al. [23, 24].

Post-Cryoplane

After the Cryoplane studies, another preliminary aircraft design was presented by Verstraete [25, 26]. He investigated a long-range application utilising two turbofans and also adopted the two-tank layout in front and aft of the passenger cabin. The design mission considered 380 passengers at a range of 7500 nmi. Similar to Brewer, he estimated energy savings of 10.4% compared with the equivalent kerosene version and an MTOW 30% lighter.

Goldberg in her PhD thesis [27], investigated the feasibility of a BWB configuration with distributed propulsion based on the architecture proposed by Felder et al. [28]. The application requirements were for 300 passengers and a

range of 7500 nmi with Jet A-1 as fuel. However, one section of the study focused on the retrofitting of the aircraft to use LH₂. Goldberg, by assessing the available volume of the airframe and the maximum fuel that can be stored, concluded that the LH₂ version had an approximately 30% reduced range. This highlights the importance of redesigning the hydrogen aircraft when comparing it with a kerosene-fuelled version of similar capabilities (design mission range and payload).

In early 2021, the Aerospace Technology Institute (ATI) conducted the FlyZero project [29] that aimed at providing a vision of carbon-free aviation by introducing a first generation of LH₂-powered aircraft with entry into service (EIS) in 2030. In an associated report [30], published in early 2022, three hydrogen aircraft concepts and three powered by sustainable aviation fuels (SAF) with equivalent design mission requirements were presented. The aircraft passenger capacity and range as well as the LH₂ to SAF difference in mission energy consumption are presented in Table 2-2. The comparison, once again, suggests that the greater the range and payload of the application the more energy-efficient the LH₂ aircraft becomes.

Table 2-2 ATI hydrogen concepts comparison with Jet A-1 [30]

Application	Regional	Narrowbody	Midsized
Passengers	72	180	279
Range	800	2400	5750
LH ₂ to Jet A-1 Energy reduction	+8%	-3.7	-6.9

ENABLEH₂

At the time of writing, ENABLEH₂ is an ongoing European Union Horizon 2020 funded project coordinated by Cranfield University that started in 2018 [31]. Its primary focus is to contribute to the advancements in LH₂ powered commercial flights and it includes partners such as Chalmers University, London South Bank University, GKN Aerospace, Safran S.A., European Hydrogen Association, Heathrow Airport and Arttic. The key areas of focus are:

1. Technology evaluation of LH₂ aircraft with EIS 2050 and comparison with equivalent Jet-A1, drop-in bio-fuels and liquefied natural gas powered applications.
2. Cryogenic fuel system and heat management through analytical and experimental work.
3. Micromix combustion addressed through analytical and experimental studies to obtain NO_x emission prediction models.
4. Safety implementation of LH₂ at aircraft, airport and operational level by characterizing and mitigating hazards
5. Roadmapping of the introduction of LH₂ in civil aviation.

One of the first activities of the technology evaluation studies was to select four LH₂ aircraft concepts, two for short to medium range (SMR) and two for long-range (LR) applications for detailed sizing. One in each range category would opt for maximum efficiency, namely “*maximum synergy concepts*” by combining advanced technologies, and the other would reflect a “*more conventional*” approach to the LH₂ aircraft. Similarly to the Brewer studies [15], several concepts with different airframe architecture, tank positioning and propulsion system configurations were qualitatively evaluated in [11, 13]. The criteria of evaluation involved four major categories such as cost per available seat kilometre, revenue, noise and safety, and their individual weighted importance, different for short to medium and long-range, was agreed upon by the partners involved.

As a result of this qualitative analysis, the maximum synergy concept selected for the SMR application was T&W aircraft, where the wing roots are extended to form a trapezoidal area to create the volume required to fit in LH₂ tanks. Its thrust is partly provided by a single turbofan mounted on the tail and a set of wing-mounted electric fans powered by fuel cells. The LR maximum synergy, namely EH2 BWB shown in Figure 1-1, is a BWB configuration with the LH₂ tanks integrated in the volume under the passenger cabin. The propulsion system features a turboelectric distributed propulsion where two semi-embedded turboshaft or turbofan engines produce the electric power required to

drive an array of fans. The fans are installed above the aft of the fuselage and ingest the boundary layer developed over it. The more conventional concepts involved T&W design with underwing mounted turbofans and LH₂ tanks mounted above the pressurised cabin for both range categories. All the selected concepts are schematically presented in Figure 2-3.

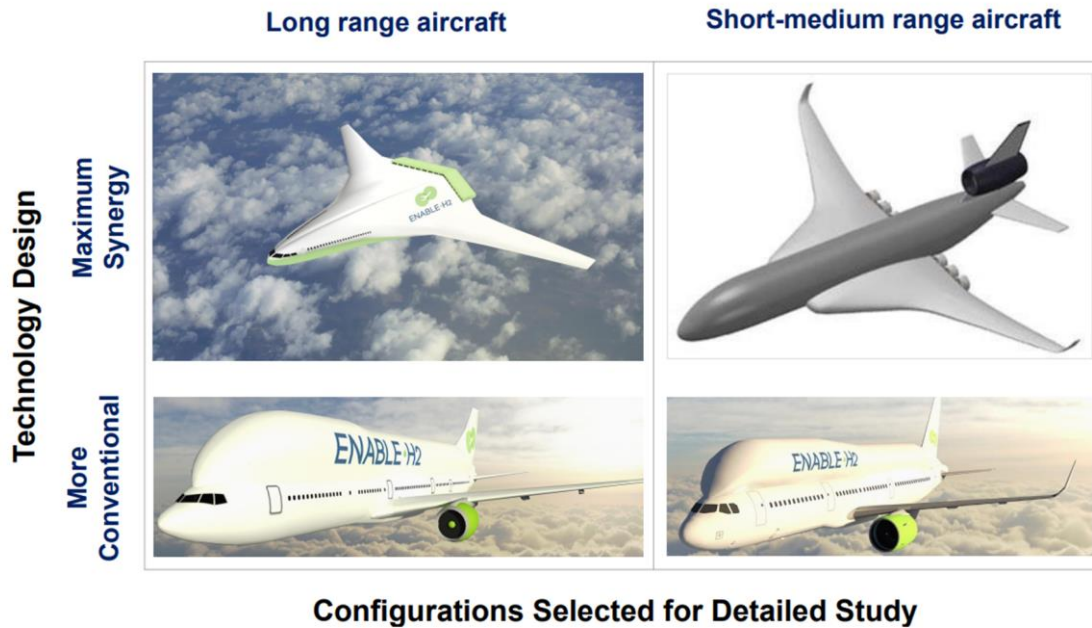


Figure 2-3 ENABLEH2 down selected configurations courtesy of [31]

Research gap

Throughout the conceptual studies presented it is clear that the majority of the concepts have a T&W airframe architecture and hitherto no application has quantified the BWB sizing for LH₂.

2.2 BWB architecture

A blended wing body airframe is a conceptual aircraft architecture proposed by Liebeck in the 1990s and was conceived as part of the advanced concept study funded by NASA [24, 32]. Contrary to the T&W configuration the fuselage smoothly transitions to the wing as part of an extended wing structure that also contributes as a lifting surface similar to a flying wing concept.

The passengers are accommodated within the lifting fuselage section in a multi-bay layout that spreads in the spanwise direction, making the fuselage shorter than an equivalent T&W. Overall this allows for a reduction in the wetted area to volume ratio providing benefits in lift to drag ratio characteristics. Also, similarly to the flying wing, the horizontal stabiliser can be eliminated, further reducing the surface area [33, 34]. Reist and Zinng [35] provided a better insight of the aerodynamic performance, by scaling an equivalent number of BWB and conventional T&W configurations with the same passenger and range capabilities. The study showcased the benefits in L/D and lift coefficient characteristics (C_L) as summarised in Table 2-3.

Table 2-3 Aerodynamic characteristics of equivalent T&W and BWB designs at start of cruise, obtained from [35]

	T&W			BWB		
Passengers	100	160	300	100	160	300
Altitude (1000 ft)	40	36	32	46	44	34
L/D	20.1	20.3	24.3	24.8	27.6	30.0
C_L	0.533	0.501	0.460	0.284	0.301	0.288

Despite the aerodynamic benefits, Liebeck also identified challenges associated with such architecture [36]. The flight controls, especially in tailless configurations, need to be multi-purposed and enable pitch, roll and yaw mechanisms. This drives the design to utilise large control surfaces installed across the span of the aircraft and, as demonstrated by Guo, potentially leads to higher noise levels [37]. A tailless configuration also poses a significant challenge to stability margins. Regarding the passenger cabin: emergency egress could also be a challenge in passenger capacities larger than 400 [32, 38] and passenger comfort may be negatively perceived as the trapezoidal cabin allows for limited access to windows.

Liebeck's proposed three-engine design estimated a 27% - 32% reduction in mission fuel compared to a four-engine double-decker T&W configuration [24, 39] as well as a 19% reduction in operating empty weight. In recognition of the

significant improvement in fuel burn several studies have been conducted [40, 41, 42, 43, 44, 45, 46, 47] in the years following the Liebeck publication. These involved sizing methodologies, weight estimation and structural analyses, noise prediction, mission optimisation, scaling studies and comparisons with equivalent T&W configurations and derivative designs as the flying-V concept proposed by TU Delft in collaboration with KLM [48].

However, apart from the airframe innovation, the unique BWB architecture allows for advanced propulsion system concepts to further improve fuel efficiency. The silent aircraft studies [42] proposed a highly embedded propulsion system where a cluster of three fans is powered by a gas turbine and three such units were installed above the aft fuselage section. The final iteration of the proposed aircraft, namely SAX-40, was estimated to reduce noise by 28%. The N+3 studies conducted by NASA [49, 50], produced the N3-X concept with EIS 2035. Considering the benefits of a highly integrated propulsion system and airframe, the N3-X features several fans installed above the aft fuselage which are powered by turboshaft engines. This configuration provides the benefits of distributed propulsion in combination with boundary layer ingestion and was predicted to further increase fuel efficiency by the propulsion system alone. Compared with a representative long-range aircraft that was in service at the time of the study, the futuristic N3-X was estimated to achieve a 60% reduction in fuel.

The BWB concept can theoretically provide increased aerodynamic performance, so much so that Airbus selected it as part of the Zero-e project [2]. In the context of this study, existing methodologies readily available in literature will be used to assess its feasibility with LH₂.

2.3 Propulsion system

Turbo-electric distributed propulsion:

Improving the efficiency of conventional turbofans is generally facilitated by reducing their specific thrust [51] and generally both lead to an increase in fan size. As aerodynamic and material limits are ever-improving throughout the years, engine core technologies are enabling larger and more efficient turbofans to be used in aviation. This increase in size, however, comes with integration limits imposed by the airframe. Such a limit is the ground clearance that restricts the diameter of the fan and therefore its maximum BPR and minimum FPR. On existing airframes, aircraft manufacturers are tackling the installation of bigger engines by relocating the integration on the wing. On clean-sheet designs where theoretically the whole aircraft can be designed for bigger engines, the increase in length and weight of the undercarriage may offset any benefits provided by the more efficient engines.

A solution to this problem can be to split one big propulsor into several smaller ones in a distributed configuration. This unconstrains the maximum diameter limits, allows for higher BPRs and introduces the opportunity for more efficient integration between the airframe and the propulsion system [52, 53].

Although distributed propulsion in its precise term has existed from the early days of aviation, in recent years the advances in electrical transmission and batteries have allowed for the introduction of novel configurations [54]. Of particular interest to this study is the turbo-electric distributed propulsion system concept introduced by Felder et al. [55] for the N3-X. Contrary to the distributed propulsion through mechanical transmission of the silent aircraft studies [42, 56, 57], the array of fans were powered by two turboshaft engines via sets of electric generators and motors. In addition to the concept proposed by Felder, several studies [27, 58, 59] have proposed a variant of this configuration the partially TeDP, where the turboshaft engines are substituted with turbofans. The turbofans partly provide thrust as well as the power required to drive the BLI fans. Both configurations are presented schematically in Figure 2-4.

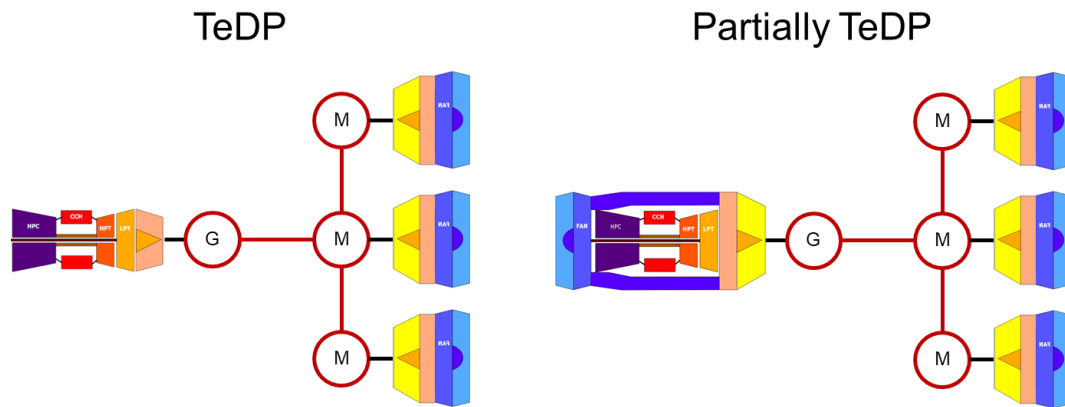


Figure 2-4 Turboelectric distributed propulsion configurations

High temperature superconducting electrical transmission system:

To obtain the efficiency benefits of the distributed propulsion, the weight and efficiency of the electrical transmission need to be comparable with that of the mechanical gearbox of a geared turbofan. As conventional electric motors and generators may not meet these standards [60], the use of superconducting technologies has been proposed as a solution [61, 62, 63]. A superconductor will allow for near-zero ohmic resistance when operating at cryogenic temperatures, which in turn enables compact and high-efficiency motors and generators [64]. Although in principle efficiencies may exceed 99%, cooling an HTS machine is a major challenge and can be achieved via a cryocooler which requires power and adds weight [65, 66]. For the N3-X transmission system, Brown [67] projected advancements in HTS technologies and estimated an overall transmission efficiency between 98% and 98.5% depending on the superconductor used. The weight of the cryocoolers was a significant proportion of the overall weight of the HTS machine. To avoid the losses associated with cooling, in a further iteration of the N3-X propulsion system, Felder et al. [28] proposed the use of LH₂ which was then burnt in the turboshaft engines in a dual-fuel capacity. This had synergistic benefits as the weight of the HTS transmission system was reduced, the efficiency increased and the Jet A-1 carried was also reduced allowing for a further improvement in fuel burn. This approach was also adopted by Kirner [68, 69, 70].

Of significant importance is the type of electrical transmission that will be selected as it needs to optimally couple the rotational speeds of the low power turbine (LPT) and the fan. The optimum performance of these components may occur at different rotational speeds and transmission ratios at different operating conditions. Jones et al. [71] compared potential configurations such as DC to DC, AC to DC and only AC between generator and motor. In that analysis, the two first candidates were found to have significant weight and efficiency losses attributed to the converters required. The full AC system was much lighter, substantially more efficient and synchronously connected the generators and motors. The optimal rotational speed of the LPT and the fan was achieved through the use of a gearbox. For the N3-X [67], it was considered to use inverters to change the current frequency to run the fan at optimal conditions, however, inverters come with power losses and additional cooling.

Boundary layer ingestion:

The potential of a boundary layer ingesting propulsion system has long been recognised [72]. A propulsor that ingests the boundary layer developed over the surface of an airframe, has the potential to re-energise its wake. Ideally, between a BLI and a propulsor in free stream air, the former requires less power for the same amount of thrust, allowing for higher fuel efficiency [73]. BLI, however, comes with the drawback of the distorted flow at the fan face that penalises its efficiency, as well as adding transmission losses if the propulsor is installed separately from the power generating unit.

To exploit the benefits of this application, several concept aircraft have been proposed. For the BWB airframe, for example, the SAX-40 [56], although the main driver of the embedded propulsion system was the noise reduction, the inlet of the propulsors were ingesting the boundary layer. In the N+3 studies, the N3-X featured 14 propulsors that re-energise almost all the boundary layer developed over the top of the fuselage. Compared with an equivalent airframe powered by geared turbofans, namely the N3A, Felder estimated an overall 41% fuel benefit at mission level [28]. This reduction however cannot be

attributed to the BLI benefit alone but also to the TeDP aspect and the potential reduction in overall propulsion system weight.

Apart from BWB applications, there are T&W concepts that feature BLI propulsors. NASA's STARC-ABL proposed by Welstead et al. [74] is such an example and is generally referred to as a "*propulsive fuselage*" application. This aircraft features two underwing-mounted turbofans that in conjunction power a single ducted fan mounted at the aft section of the fuselage. The estimated design mission fuel saving compared with a conventional twin turbofan configuration was estimated at 12%. Approximately this level of fuel efficiency was also predicted by the CENTRELINE project [75]. Last but not least is the D8 concept proposed by Drela [76]. It is a short-range application that features a double-bubble fuselage that also acts as a lifting surface. It is powered by two BLI turbofans that are mounted at the back of the fuselage under a "π" shaped empennage.

N3-X TEDP sizing and subsequent studies:

Based on the N3-X platform several studies tackled aspects of its TEDP system. Liu et al. [77] presented a methodology to assess the performance of a BLI fan due to inlet flow distortion. Valencia and Laskaridis [58, 59] performed a design space exploration by addressing a few design parameters such as FPR (similarly to [55]) and the number of fans. Goldberg et al. [78, 79] introduced a method to estimate the boundary layer control volume for the propulsion system sizing and optimised the design parameters of the BLI fan array. Athanasakos et al. [80] suggested a model to account for both distortion effects, utilizing the parallel compressor theory, and reduced inlet momentum drag due to BLI. Across the design space exploration from the studies mentioned [27, 55, 59, 78, 80], it was determined that low fan pressure ratios and larger numbers of fans allow for increased propulsion system efficiencies. Valencia [59] also estimated that a partial TeDP configuration, where the turboshaft is replaced by a turbofan, might potentially lead to overall higher efficiency.

Research gaps

Previous conceptual studies [20, 25, 26] on LH₂ aircraft were mainly focused on conventional turbofan designs and engine design parameters were selected based on engine efficiency. The TeDP concepts discussed focused on optimising the performance of the fan array but considered constant cycle parameters (i.e. overall pressure ratio and turbine inlet temperature) for the sizing of the turboshaft. In addition, for the partially TeDP the sizing of the turbofan as well as its thrust split with the BLI fan array have only partly been addressed in the past. Finally, although potential benefits of a synchronous superconducting electrical transmission system have been recognised, their implementation with variable area nozzle on the BLI propulsor has not been addressed.

Based on the above the following research questions are derived:

How would the design parameters of a LH₂ TeDP and a partially TeDP be selected based on mission fuel efficiency?

How would a multipoint design approach affect the overall sizing of a TeDP configuration?

How would a synchronous electrical transmission system coupled with VAN drive the rotational speed selection and weight estimation of the HTS components?

2.4 LH₂ tanks for aircraft applications

Storage conditions

There are a few ways that have been proposed in the past to effectively store hydrogen either in pure form or as part of a more complex substance. For the latter, as summarised by Zuettel [81] and Felderhoff et al. [82], hydrogen can be carried on board in a compact metal hybrid form, such as Mg₂FeH₆, and then extracted under atmospheric conditions through catalytic desorption. Hydrogen can also be extracted through hydrocarbon reforming [83], however, both options come with a substantial weight penalty and therefore are not preferable for aircraft applications.

Considering hydrogen properties there are several conditions under which hydrogen can be stored and achieve a relatively high density. A few representative states are summarised in Table 2-4 extracted from the National Institute of Standards and Technology (NIST) [84]. As adopted by the automobile industry, hydrogen can be stored under ambient temperature in a supercritical state but needs to be highly compressed to even allow for relatively high density. These storage conditions are non-preferred for large quantities applicable to medium and long-range aircraft but are potentially viable solutions for small air mobility applications. An additional advantage of this option is that it does not require the complex fuel distribution and heat management system that is required for the cryogenic storage options listed below.

At a pressure of 1 bar the boiling point of hydrogen is 20.2 K, if stored at temperatures below that, in a subcooled state, it would allow higher density than that at its boiling point. However, the increased energy required to cool it to a lower temperature outweighs the benefit of a slight increase in density. At the same time, to maintain hydrogen at 1 bar in the subcooled state either variable tank volume or a pressurising medium is required. During the Bee project [6] this was tackled by using helium at cryogenic temperatures to pressurise the tank but in the long term, this is not a sustainable solution because of the limited availability of helium.

When stored at its boiling point the liquid and gaseous state co-exist in the tank. As LH₂ is withdrawn, the GH₂ occupies the remaining space effectively acting as the pressurising medium. The disadvantage of this option is that due to heat ingress in the tank the LH₂ evaporates and builds up pressure. To safely store it under such conditions pressure control mechanisms need to be considered as proposed by Lin [85]. Regardless of this aspect, LH₂ at saturated conditions is most commonly selected throughout the conceptual studies reported in section 2.1. The last option considered is to store hydrogen at 350 bar and approximately 34 K at a supercritical state. There it has the highest density compared to the other options examined. However, this combines the disadvantages of both cryogenic temperatures and high pressures leading to a heavy solution.

Table 2-4 hydrogen properties at different possible storage conditions

Condition / State	T (K)	P (bar)	ρ (kg/m ³)
Subcooled / Liquid	15	1	76.1
Saturated / Liquid and Gaseous	20.2	1	70.8
Cryo-compressed / Supercritical	33.8	350	85
Compressed / Supercritical	288	700	40.2

Types of LH₂ tanks

A cryogenic tank needs to serve a double purpose: limit the heat leak from the environment and support the pressure developed over time. The former is tackled by the insulation and the latter by the wall.

Insulation:

Regarding the insulation, for LH₂ storage on aircraft applications three main design options have been practically and theoretically considered, as summarised by Khandelwal et al. [86], vacuum, multilayer and foam insulation:

1. Vacuum with near-zero thermal conductivity, allows for low heat ingress but comes with significant tank weight and may lead to catastrophic

failure upon loss of vacuum [87]. For long term LH₂ storage, this method has been widely used if weight is not the main driver but might also have potential in small aircraft applications.

2. Multilayer insulation (MLI) utilises several layers of thermal radiation shields that consist of alternate thin layers of metal foil and insulating material. Its thermal conductivity depends on near-vacuum pressure of the residual gas in the insulation. It is a very compact and overall lighter solution than vacuum and has low thermal conductivity but not as low as vacuum. It is costly to implement and maintain and also loss of vacuum may lead to catastrophic failure. Verstraete et al. [88] and Huete et al. [89] demonstrated that MLI can potentially be a relatively light option for smaller tanks applicable to small aircraft.
3. Rigid polymer foam insulation has lower density than vacuum and MLI but higher thermal conductivity. It is a well-established option for its low cost and ease of implementation but comes with the challenges of high heat ingress to the tank contents. For large tanks in aircraft applications, it can provide the lightest solution overall and is widely selected throughout conceptual studies, [5, 20, 25, 88, 90]. It was also used for LH₂ tanks of the Boeing® Phantom Eye long-endurance UAV [91, 92].

Alternatively to foam insulation, aerogels are known for their very low thermal conductivity and low density and may be a foam replacement but they are not yet well established and have limited mechanical properties [87].

Tank wall:

Regarding the wall, it is most commonly integrated inside the foam [20, 91, 92, 93] to protect and seal the insulation from the cryogenic fuel (hydrogen permeation in the foam may lead to increased thermal conductivity and loss of material integrity [87]). The wall material needs also to resist hydrogen permeation and embrittlement [20, 81, 94]. Aluminium alloys are commonly selected for their properties under cryogenic conditions. Composite materials are a lighter option but since they are generally prone to hydrogen permeation a protective liner should additionally be used [95].

Tank shape:

For his configuration, Brewer [18, 20], regarding the aft tank shown in Figure 2-1, considered either high integration with the airframe or none and concluded that the former would be a more efficient design both gravimetrically and volumetrically. Its shape, dictated by the geometry of the rear part of the fuselage, was conical with ellipsoidal cap ends. The front tank was spherical. Spherical tanks offer better surface to volume characteristics than cylindrical tanks for the same content capacity and withstand the same pressure loads with thinner walls [96, 97] but volumetrically are less efficient. For aircraft applications, the main driver is to store as much fuel as possible by simultaneously occupying the least space and allow for light weight.

Research gaps:

In the context of this work, LH₂ tanks need to be sized. The tank sizing methodologies reported in the literature [89, 98, 99] are only considering a single set of conditions for the LH₂ stored, usually when the tank is full. In reality, however, the conditions that affect the heat transfer are ever varying throughout a flight. Brewer [20] and Verstraete et al. [88] accounted for that effect and estimated the performance of an LH₂ tank at mission level. The former, for a fixed geometry, demonstrated how much GH₂ needs to be extracted throughout a mission to avoid overpressure. The latter showcased how the maximum pressure can optimally be selected to avoid GH₂ extracting for varying tank sizing and mission requirements. Although sizing the LH₂ tank for a higher pressure to avoid GH₂ venting seems a reasonable strategy, sizing it with active pressure control for lower pressures would lead to a lighter design and is yet to be quantified. In addition, literature widely reports the effect of LH₂ evaporation due to heat ingress in the tank, however, no studies to date have investigated how to address the sizing to compensate for the evaporation effect.

Based on the above observations the following two research questions are placed:

Are there potential weight savings when GH_2 extraction for pressure control is considered?

How can the unusable LH_2 be quantified over a flight profile and how does it affect the tank sizing process?

3 Methodology

Contents:

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3.4	Propulsion system sizing.....	62

This chapter includes a detailed description of the individual modules used as well as their integration in the hydrogen aircraft sizing framework developed. The modules are aircraft performance and weight estimation, LH₂ tank sizing, and propulsion system performance and weight estimation. At first, a generic overview of how these are integrated into the framework is introduced and how they function co-dependently. Then in dedicated sub-sections, each module is thoroughly presented. For the numerical tools of these modules, both developed and existing ones were utilised. The use of these tools was equally based on newly developed methodologies as well as existing and adapted ones. Finally, the models were mostly developed and in some cases adapted from existing ones. A summary of these is presented in Table 3-1.

Table 3-1 Summary of tools, methodologies and models used

		Tool	Sizing Methodology	Models
Aircraft	Performance	E	E	D / A
	Weight	E	E	D
LH₂ tanks	Performance	D	D	D
	Weight	D	D	D
Propulsion system	Performance	E / D	A / D	D
	Weight	E / D	E	D
Integration Framework		D	A	-

E: Existing D: Developed A: Adapted

3.1 Integration framework

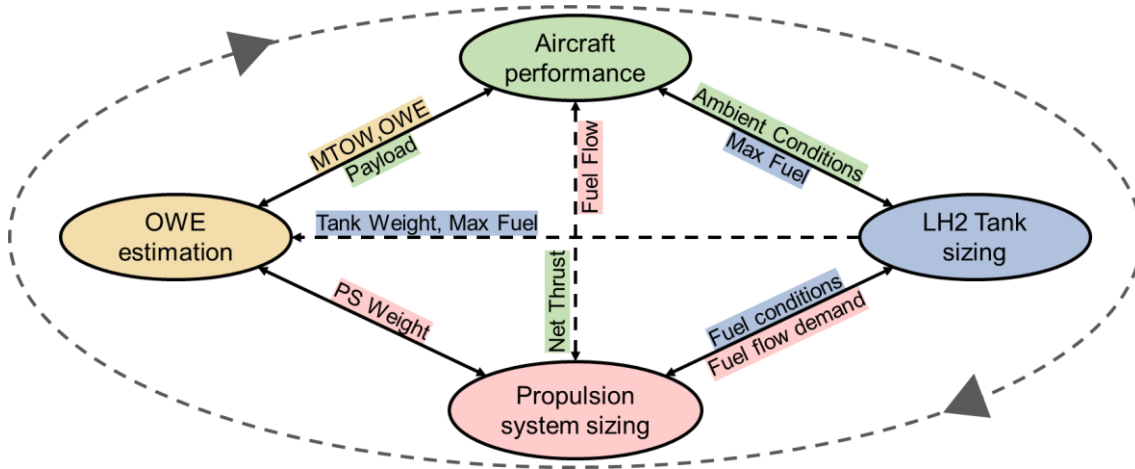


Figure 3-1 Hydrogen aircraft integration framework and module dependencies

As briefly discussed the hydrogen aircraft framework developed consists of four distinctive modules, LH₂ tank sizing (LTS), propulsion system sizing (PSS), aircraft weight estimation (AWE) and aircraft performance (AP). Schematically these are presented in Figure 3-1 together with the most significant parameters communicated between them.

LH₂ tank sizing: Optimally sizes LH₂ tanks for a given aircraft geometry. Requires ambient conditions from the mission profile from AP, and fuel flow demand from PSS.

Propulsion system Sizing: Creates models for propulsion system performance and estimates its weight. Requires fuel conditions from LTS for fuel pre-heating considerations and thrust requirements from AP for the sizing as well as performance prediction.

OWE estimation: Estimates the aircraft OWE and MTOW. Requires the weight of the tanks and propulsion system predicted from LTS and PSS respectively as well as the design and maximum payload considered in AP.

Aircraft performance: Estimates the fuel burn across a mission profile. For a provided aircraft geometry, requires the MTOW, OWE predicted by AWE, fuel flow demand of the propulsion system by PSS and the maximum fuel capacity from LTS.

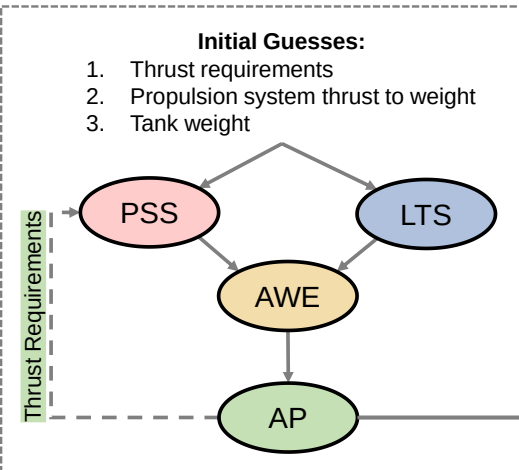
As the majority of the shared parameters between the modules are required for the computation of the others, several iterations may be performed between them until the results of each process reach a converged solution. However, to avoid complexity when the framework is used, a simplified approach is followed that focuses on the LH₂ tank sizing optimisation as well as propulsion system design space exploration.

As a starting point, the aircraft geometry is sized based on the requirements of the passenger cabin as well as tank available space. At the same time the propulsion system is sized based on estimated thrust requirements as well as power to weight ratios and the tanks according to pre-set gravimetric efficiencies. Based on the above the weight of the aircraft is estimated and its design mission performance calculated. Several iterations are then performed to correct for the calculated thrust levels as well as fuel volume that would match the design mission requirements. Once the initial sizing is finished, the tanks are optimally sized in detail based on the available space, fuel flow scheduling and ambient conditions and their weight as well as maximum fuel capacity are used to resize the aircraft. At the same time the design space exploration for the propulsion system is performed for the thrust levels determined by the initial sizing. For every set of propulsion system design parameters a different aircraft is then sized. The optimum propulsion system configuration is then selected based on the design that achieves the longest range for the maximum LH₂ carried by the tanks. It is noted that during the design space exploration, both the thrust requirements as well as the tanks are not reconfigured based on the latest sizing iteration. The sizing methodology is schematically presented in Figure 3-2. Part of this framework was used by the author to size a T&W aircraft powered by liquefied natural gas [14].

Starting Point:

Aircraft design mission requirements
Aircraft geometry

Initial sizing:



LH₂ tank optimisation :

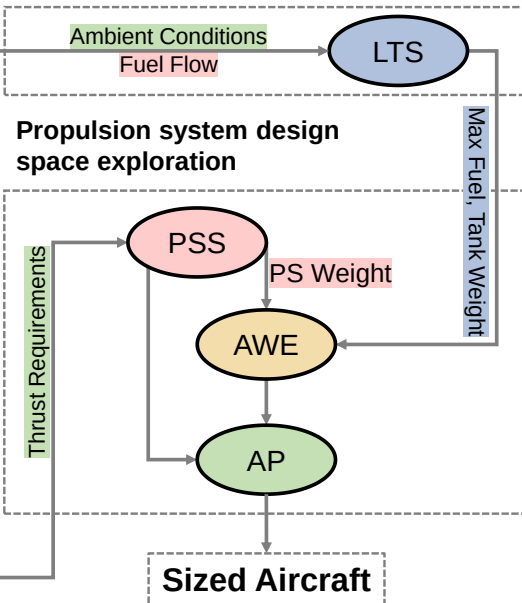


Figure 3-2 Hydrogen aircraft design methodology for tank sizing optimisation and propulsion system design exploration

3.2 Aircraft sizing & mission analysis

In this section the aircraft modelling tools are discussed, the aircraft performance and the weight estimation modules.

3.2.1 Aircraft performance

The tool used for the aircraft performance is an in-house code developed within Cranfield University called Orion, that couples aircraft drag calculations of a given airframe with propulsion system performance. A thorough description of the tool, its functionality and its limitations is provided by Goldberg and Nalianda [27, 100]. The key functions of the aircraft performance model are as follows:

3.2.1.1 Aircraft drag

The aircraft drag module estimates the drag of an airframe according to a defined geometry and the mission profile (i.e. speed and altitude) established by the mission trajectory definition module described below. The drag estimation is based on a single point approach, where all the force components that act on the aircraft are assumed to be applied at its centre of mass. The drag of the airframe is considered to consist of three components: profile drag, lift-induced drag, and a transonic parasitic drag term. Aircraft planforms are approximated using basic shapes to simplify drag prediction.

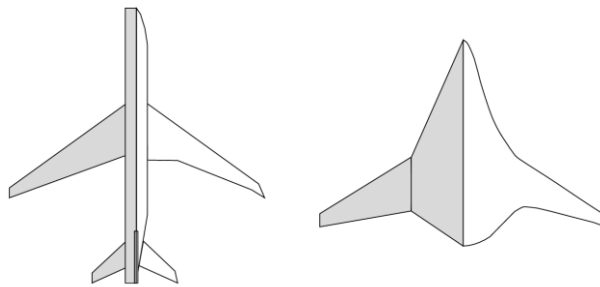


Figure 3-3 Simplified planforms for a tube and wing (left) and a blended wing body (right) airframe [27].

Profile Drag: Airframe profile drag is estimated using a drag build-up method which combines all the aircraft components to predict the total profile drag, such as wings, fuselage, horizontal and vertical tail, nacelle, undercarriage and high-lift systems. The total airframe profile drag is obtained by the summation of

individual component profiles which in turn are estimated by empirical relationships, detailed by Jenkinson et al. [101] and Raymer [102].

Lift induced drag: Lift induced drag is produced by the lifting surfaces alone. For a BWB application, since lift is produced by the fuselage section as well, the airframe is considered to be a flying wing with an equivalent planform area and aspect ratio of the fuselage and wing area section. If winglets are included in the architecture a correction is applied to the aspect ratio of the wing [103].

Transonic Parasitic Drag: Transonic wave drag components are introduced when the flight Mach number exceeds the aircraft's critical Mach number (the Mach number for which shocks occur and begin to influence drag). It is estimated graphically as introduced by Raymer [102].

3.2.1.2 Mission profile definition

A mission is broken down into flight phases each consisting of several segments as follows:

1. Start-up and Taxi-out
2. Take-off and initial climb to 1500 ft
3. Climb from 1500 ft to cruising altitude
4. Cruise in a steady-level flight at a specified speed and altitude.
5. Descent from cruising altitude to 1500 ft.
6. Landing
7. Diversion and Hold

For every segment of the flight, velocities, altitude and ambient conditions are pre-set by the user.

3.2.1.3 Integration with propulsion system and mission performance

Integration with propulsion system simulation software:

To predict the engine performance of every segment, Orion is integrated with an in-house tool for gas turbine performance simulation: Turbomatch. For every segment of the flight the ambient conditions, altitude, flight speed and a performance control parameter are transferred to the engine performance tool

and the engine performance is predicted. The control parameter for the engine can be the net thrust or the engine power setting. At cruise conditions, where steady flight is assumed, the aircraft drag is predicted and is considered equal to the total engine thrust and the thrust, in turn, is used as a direct input. During the accelerating and decelerating phases of the flight, an engine parameter that dictates the engine power setting is used such as the shaft rotational speed, turbine inlet temperature, engine fuel flow or another. As a result, the rate of climb or descent of the aircraft is calculated. The engine simulation software then provides information about the engine performance with the most significant one being the engine fuel flow.

In the context of this study, since advanced propulsion system configurations are investigated, another gas turbine performance simulation software is integrated with the aircraft performance tool: PROOSIS. As PROOSIS and Turbomatch function with the same principles discussed by Kurzke [104], minor changes have been introduced to allow Orion and PROOSIS exchange information in exactly the same way as Turbomatch does.

Mission performance:

For the mission performance calculations, at each segment, the fuel flow is estimated based on the propulsion system performance and the flight conditions. Then for the immediate next segment, the aircraft weight is adjusted to account for the fuel burnt in the previous one. The process is repeated for the whole flight envelope and block mission fuel is calculated as the sum of the fuel burn of all the individual segments. On top of the block fuel, reserves are considered as well based on diversion and hold profiles and an additional 5% of the block fuel is included for contingency. The onboard fuel is the sum of all discussed fuel components as dictated by eq. (3-1).

$$M_{onboard} = M_{block} (1 + contingency) + M_{diversion} + M_{hold} \quad (3-1)$$

The main outputs required from the model are block time, fuel burn, and CO₂ emissions. The model can simulate a mission with either a defined range, a

defined take-off weight, or a given trajectory. In addition, the model can produce a payload-range chart.

In the tool, two distinctive calculation modes exist for mission performance analysis, one that uses the mission range as an input and another that uses the take-off weight of the aircraft. Both of these modes use as inputs the OWE and the mission payload.

- **Target range:** As at the beginning of the mission the aircraft weight is unknown, an initial estimation of the fuel weight needs to be provided. This is performed via a derivation of the Breguet Range Equation shown in eq. (3-2) [102] according to the range of the mission (R), the specific fuel consumption of the engines (SFC), the cruise flight velocity (V) and the lift to drag ratio at cruise (L/D). Based on the fuel consumption calculated in the first iteration, the take-off weight of the aircraft is adjusted and the mission is calculated from the beginning. The process repeats until the mission fuel calculated between iterations shows user acceptable agreement.

$$M_{fuel, cruise} = \exp\left(\frac{-R \times SFC}{V \times (L / D)}\right) \quad (3-2)$$

- **Target take-off weight:** Contrary to the target range mode, at the beginning of the mission the fuel weight is known. However out of this quantity the split between the usable fuel and the reserve fuel is unknown. Similarly to the other mode, several iterations are performed until the initial guess of the usable fuel supports a mission with the target overall mission fuel requirement.

3.2.2 Aircraft weight estimation

The weight estimation module is based on the method developed by NASA [105] (integrated with the Flight Optimisation System (FLOPS)) and is used instead of the methodologies of Raymer [102], Torenbeek [106] and Roskam [107] due to its capability to estimate the weight of a BWB architecture. This method includes different equations according to aircraft type (i.e. civil aviation T&W and BWB, fighter aircraft and general aviation) for all the components contributing to the OWE. Contrary to T&W, the BWB are as yet conceptual aircraft, and as such the wing and fuselage weight estimation correlations are derived from finite element analysis studies and are comprehensively described in [105].

Validation of the tool has been performed by Horvath et al. [108] on a 737-200. The weight estimates of several components of the aircraft are compared against the actual data found in [107], providing a total OWE error of just -2.1%. The study concludes that the method adequately captures overall trends in component weight and provides a reasonably accurate prediction of component weight. In addition, it is noted that although this weight estimation method and the ones of Raymer [102], Torenbeek [106] and Roskam [107] present some similarities, it is difficult to compare them due to differences in component groupings and lack of thorough documentation of the data sets used in their creation.

Validation of the tool is also performed in this study on six T&W aircraft of different sizes and range capabilities, as well as a long-range conceptual BWB configuration. The airframe data and geometries were derived from the corresponding aircraft characteristics for airport planning documents [109, 110] and each engine weight was used directly as an input. For the BWB, data from [28] are used together with the geometry reported in [111]. Although this is a conceptual aircraft and its weight is estimated in [28] by the same methodology, it is included here as well to ensure the validity of the results. The validation results are presented in Table 3-2 in terms of predicted OWE and its component breakdown. In all cases and due to lack of detailed information, only the

estimated OWE could be compared against the manufacturer quoted values. The tool presents relatively small prediction errors.

Table 3-2 Weight estimation tool validation (tonnes)

Aircraft	737-200	737-800	767-200	777-200LR	A320-200	A340-200	N3A
Structure total	14.1	20.2	41.6	91.2	20.8	70.8	80.1
Propulsion total	3.6	6.0	11.3	22.4	6.1	13.8	13.1
Systems and Equipment	8.6	12.0	16.7	25.0	11.9	24.4	29.2
Operating items total	1.5	2.5	4.9	7.1	3.1	7.2	5.7
Predicted OWE	27.9	40.7	74.6	145.7	41.9	116.2	128.2
Actual OWE	28.4	41.4	79.0	146.0	42.6	117.9	129.3
Prediction Error	-1.90%	-1.74%	-5.54%	-0.26%	-1.62%	-1.48%	-0.85%

3.2.3 BWB cabin sizing

The passenger cabin of a T&W aircraft is driven by the number of passengers and the seating arrangement for both economy and first class. The total length and area of a single bay can be trivially estimated by including typical dimensions for the passenger seating, lavatories galleys and closets. On a BWB passenger airliner, however, the cabin is separated into several shorter bays as proposed by [32]. The cabin area and dimensions are then estimated based on both the number of bays and the passenger count for each class according to the methodology introduced by [112] and were later included in the weight estimation module [105]. A schematic representation of two cabin arrangements with different numbers of bays is shown in Figure 3-4.

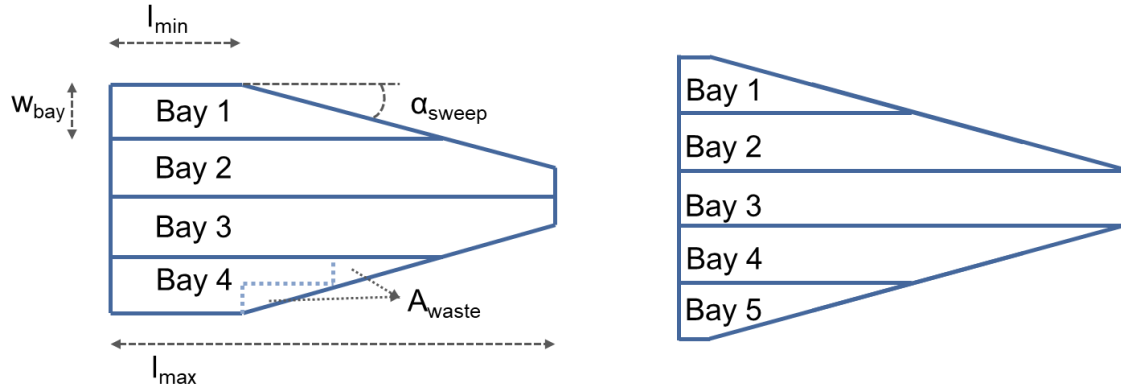


Figure 3-4 BWB cabin layout

In more detail, to calculate the dimensions of the cabin, first, its area is estimated based on the requirements for the seating, lavatories, closets and aisles, eq. (3-3). The width of a single bay (w_{bay}) is determined by the seating arrangement of the economy class. The seating area ($A_{seating}$) is a function of the passenger count. The area required for lavatories ($A_{lavatories}$) is also a function of the number of passengers as well as the width of each bay. The closets galleys and aisle areas together with the unusable cabin space (A_{waste}) are calculated as in [105]. The cabin lengths ($l_{min,cabin}$ and $l_{max,cabin}$) are then determined by eq. (3-4) and eq. (3-5).

$$A_{cabin} = A_{seating} + A_{lavatories} + A_{closets} + A_{galleys} + A_{aisle} + A_{waste} \quad (3-3)$$

$$l_{max,cabin} = \frac{A_{cabin}}{w_{bay} n_{bay}} + \frac{w_{bay} n_{bay}}{4} \tan(a_{sweep}) \quad (3-4)$$

$$l_{min,cabin} = l_{max,cabin} - \frac{w_{bay} n_{bay}}{2} \tan(a_{sweep}) \quad (3-5)$$

3.3 LH₂ tank sizing

In this section, the hydrogen tank sizing methodology is analysed and a novel way of optimally selecting the tank dimensions and materials based on mission performance is introduced. This methodology and its key outcomes are discussed in the peer-reviewed conference paper produced by the author [12] and are included herein with some subsidiary material.

At first, to provide a better understanding of the behaviour of the LH₂, the stored hydrogen properties are discussed. Then the mechanical and thermal sizing is presented and is followed by the multipoint performance assessment. Finally, an on-flight application case is provided to demonstrate the optimum dimensioning and material selection processes, both for tanks that do not vent GH₂ for pressure control and for ones that do.

3.3.1 Stored hydrogen properties

For LH₂ storage the hydrogen is considered to be at boiling conditions. At any given time when there is LH₂ in the tank, the remaining space is occupied by coexisting GH₂. Two isomers of hydrogen can be found, orthohydrogen and parahydrogen, depending on the spin of the protons. For the tank sizing, parahydrogen properties are used since at the normal boiling point the equilibrium composition is 99.8% parahydrogen [113].

Table 3-3 summarises the properties of hydrogen both for the liquid and gaseous state at indicative temperatures and pressures. At 1 bar, the boiling point is 20.2 K and the liquid and gaseous phases have densities of 70.9 and 1.32 kg/m³ respectively [84]. For saturated LH₂ the lower the pressure, the lower the temperature and the higher the density. On the contrary, for saturated GH₂ the lower the pressure and temperature the lower the density. Although the gaseous density is increasing with increasing pressure, since the liquid is always denser than the gas, it would be more volumetrically efficient not only to store hydrogen as a liquid but at the lowest possible pressure. Naturally, there is a limit as well, as although at pressures below 1 bar the LH₂ is denser, it also needs to be cooled below 20 K. The energy required to achieve this makes it

not an attractive option. Additionally, the sub-atmospheric pressure makes it difficult to handle and risks tank implosion or the entry of air. Storing LH₂ at above ambient pressure and in saturated conditions together with GH₂ is a safer option. The tank contents can remain a mixture of liquid and vapour throughout the mission, and initially, both phases can be assumed to have homogeneous pressure and temperature.

Table 3-3 Hydrogen properties for saturated LH₂ and GH₂

Temperature (K)	Pressure (bar)	LH ₂ density (kg/m ³)	GH ₂ density (kg/m ³)
18.1	0.5	73.15	0.71
20.2	1.0	70.88	1.32
27.1	5.0	60.79	6.13
31.3	12.86	31.32	31.32

Apart from the individual phase properties, it is important to consider the mixture properties. The T-s diagram of the saturated conditions is shown in Figure 3-5 [84]. The state of the mixture can be fully defined according to two parameters:

- The first is either the pressure or temperature which in turn will define the densities (ρ), latent heat of vaporization (h_{lg}), thermal conductivities (k), specific heat capacities (C_p) etc. individually for the liquid and gaseous state (vertical position on the T-s diagram).
- The second parameter is the mixture composition. This can be defined by the quality (x) (ratio of vapour mass to mixture mass) or the vapour to liquid volume or mass ratios which define the density, enthalpy and entropy of the mixture as a whole (horizontal position on the T-s diagram).

Finally, the absolute quantities of the mixture, can be calculated by providing one parameter for the mass or volume of the mixture or for the liquid or gaseous phase. These parameters are summarised in Table 3-4.

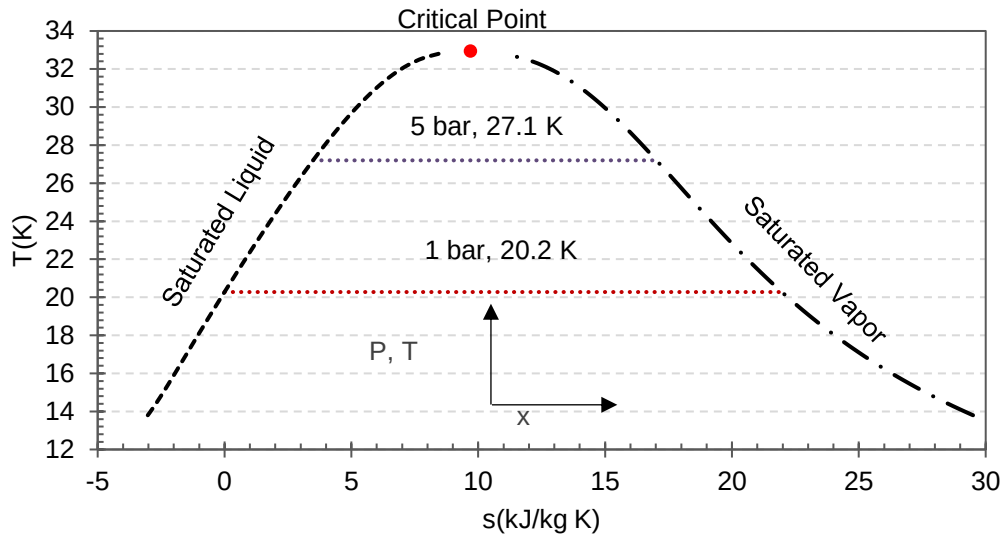


Figure 3-5 T-s diagram for liquid and gaseous hydrogen in saturated conditions

Table 3-4 Hydrogen mixture properties and defining parameters

	Properties	Parameters
Individual phase properties	ρ , s , C_P , k , h_{lg}	P , T
Mixture properties	ρ_{mix} , h_{mix} , s_{mix}	x , V_{GH2}/V_{LH2} , M_{GH2}/M_{LH2}
Quantities	V_{mix} , V_{GH2} , V_{LH2} , M_{mix} , M_{GH2} , M_{LH2}	V_{mix} , V_{GH2} , V_{LH2} , M_{mix} , M_{GH2} , M_{LH2}

3.3.2 Mechanical and thermal sizing

3.3.2.1 Geometry and mechanical sizing

The geometry considered for the tanks is cylindrical with hemispherical cap ends. The dimensions then can be trivially calculated using related shell geometries. For given insulation and wall thickness, the overall dimensions are defined according to any two of the following parameters:

1. Total mass of the mixture stored (M_{mix}), which according to the mixture density defines the internal volume (V_{mix}).
2. The length of the cylindrical section (l).
3. The total tank length (l_{tot}).

4. The external diameter (D_2).

A representation of the tank layout is presented in Figure 3-6.

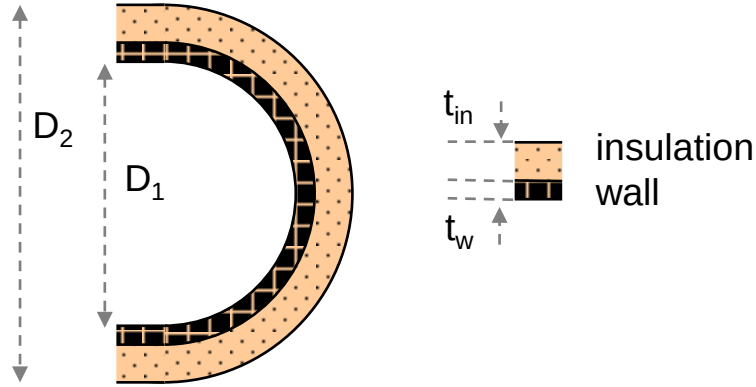


Figure 3-6 Schematic of the tank structure

The wall material needs to combine a high yield strength under cryogenic conditions and low density. A NASA study [16] on an LH₂ fuelled subsonic aircraft, and then Brewer [20], suggested the use of Al 2219 for its manufacturability and performance at cryogenic temperatures. It is also used in this study. However, other recent studies suggest the Al-Cu-Li alloy Al 2195 might be a better choice on account of its higher strength and lower density [114].

As discussed by Barron [115], the wall thickness is determined according to the ASME Code, Section VIII by equations (3-6) and (3-7) for the cylindrical part and the hemispherical cap ends respectively. The thickness is a function of the inner tank diameter (D_1), the maximum allowable pressure difference (ΔP), a factor of safety for the pressure (FoS), the maximum allowable stress of the material (σ_a) and the weld efficiency (e_w). The mass of the wall is then calculated based on the shell geometry and the density of the material.

$$t_w = \frac{D_1 \times \Delta P \times \text{FoS}}{2\sigma_a e_w - 1.2 \times \Delta P \times \text{FoS}} \quad (3-6)$$

$$t_w = \frac{D_1 \times \Delta P \times \text{FoS}}{4\sigma_a e_w - 0.4 \times \Delta P \times \text{FoS}} \quad (3-7)$$

The FoS accounts for relief valve tolerance, ultimate loads due to manoeuvre or gusts and burst pressure limits. A list of the inputs used for the materials is presented in Table 3-5.

Table 3-5 Material related wall sizing parameters

Parameter	Units	Value
Maximum allowable stress, σ_a [20]	MPa	172.4
Factor of Safety, FoS [20]	-	2.2
Weld efficiency, e_w [115]	-	0.9
Density	kg/m ³	2840

3.3.2.2 Thermal modelling and heat transfer

Several studies in the past have proposed heat transfer models for estimating the heat leak of LH₂ tanks. Reynolds [116] and later on Colozza [96] and Huete [89] estimated the heat transfer for foam insulated tanks by equalising the heat transfer by convection ($Q_{\text{conv,air}}$) and radiation ($Q_{\text{rad,air}}$) from the ambient air to the outer tank surface with the heat transfer via conduction (Q_{cond}) through the insulation of the wall. Verstraete et al. [88] and Winnefeld et al. [99] included the heat transfer via convection from the liquid ($Q_{\text{conv,LH}_2}$) and gaseous phase ($Q_{\text{conv,GH}_2}$) of the hydrogen to the inner surface of the wall as shown in equation (3). Neglecting fuel convection leads to a small overestimation of the heat transfer as the thermal resistance between the fuel and the inner wall is not considered.

In this context, the heat transfer model described by equation (3-8) is used and is presented schematically in Figure 3-7, where T_f is the temperature of the fuel stored, T_{s1} the internal wall temperature, T_{s2} the external wall temperature and T_{amb} the ambient temperature. The following assumptions are also made:

- The hydrogen mixture is homogenous, hence temperature and pressure of the gaseous and liquid states are the same everywhere inside the tank, neglecting the effects of temperature stratification and gravity.
- Heat transfer occurs under steady-state conditions.

- Thermal and pressure expansion of the tank is neglected and therefore the tank is considered to have constant volume.
- The thermal resistance of the aluminium wall is discounted as it is orders of magnitude lower than that of the insulation.

$$Q_{conv,LH_2} + Q_{conv,GH_2} = Q_{cond} = Q_{conv,air} + Q_{rad,air} \quad (3-8)$$

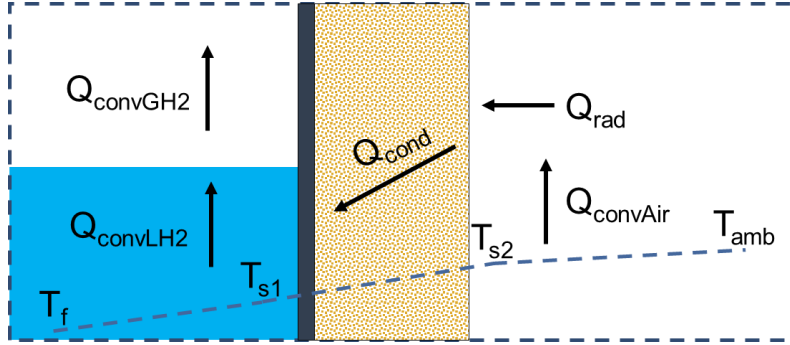


Figure 3-7 Schematic of the tank wall heat transfer

The fuel convective heat transfer ($Q_{conv,LH_2} + Q_{conv,GH_2}$) is estimated according to the convective heat transfer coefficient, the corresponding wetted surface area of each phase and the inner wall, and fuel temperature, eq. (3-9). The convective heat transfer coefficient (h) is a function of the thermal conductivity (k) of each phase, the characteristic length (l) as well as the Nusselt number (Nu), eq. (3-10). For the liquid phase, the Nu is estimated as a function of the Rayleigh number according to the empirical correlation proposed in [117], eq. (3-11). In turn, the Rayleigh (Ra) number is calculated based on the gravitational acceleration (g), the thermal expansion coefficient (β), the temperature difference, the fill level (l_{fill}), the Prandtl Number (Pr) and the kinematic viscosity (ν), eq. (3-12). All these properties can be found in [84]. The Nu of gaseous phase is set to 17 as [20], as there is limited information on its estimation.

$$Q_{conv,LH_2} + Q_{conv,GH_2} = (h_{LH_2} \cdot S_{LH_2} + h_{GH_2} \cdot S_{GH_2}) (T_{s1} - T_f) \quad (3-9)$$

$$h = \frac{Nu \cdot k}{l} \quad (3-10)$$

$$Nu_{LH_2} = 0.0605 \cdot Ra^{1/3} \quad (3-11)$$

$$Ra = \frac{g \cdot \beta (T_{S_1} - T_f) l_{fill}^3 Pr}{\nu} \quad (3-12)$$

The heat transfer via conduction (Q_{cond}) is estimated based on the thermal resistance of the insulation (R_{cond}) and the temperature difference between the surfaces, eq. (3-13). The latter is calculated differently for the cylindrical part and the end caps and depends on the tank shell dimensions and the thermal conductivity of the insulation [88, 96, 99].

$$Q_{cond} = \left(\frac{T_{S_2} - T_{S_1}}{R_{th}} \right) \quad (3-13)$$

The air convective heat transfer is a function of the convective heat transfer coefficient (h_{air}), the external surface area (S_2) and the temperature difference between the ambient air and the surface of the tank, eq. (3-14). The convective heat transfer coefficient can then be determined based on natural or forced convection correlations depending on the tank integration with the airframe. For example, if the tanks are installed in an enclosed space then natural convection can be considered, if part of the tank is an integral part of the airframe then forced convection should be used. Finally, heat transfer by radiation can be estimated based on the Stefan Boltzmann constant, the external surface emissivity and the temperature difference, eq. (3-15).

$$Q_{conv,air} = h_{air} \cdot S_2 (T_{amb} - T_{S_2}) \quad (3-14)$$

$$Q_{rad} = \sigma \cdot \varepsilon \cdot S_2 (T_{amb}^4 - T_{S_2}^4) \quad (3-15)$$

For the system of equations described, (3-8) to (3-15), the heat transfer coefficients are a function of the temperatures. Fuel ambient conditions can be used as inputs, however, to fully define the surface temperatures, they need to be initialised and iterated until their combination satisfies the heat balance of equations (3-8).

3.3.2.3 Pressure rise estimation

The heat transfer model described estimates the heat flux of the tank based on (among other parameters), a set of ambient and fuel temperatures. However, ambient temperature changes with altitude and fuel temperature with the heat input. Estimating the pressure variation is of primary importance as it would set the requirements for maximum allowable pressure difference (ΔP), used to size the tank wall.

Based on the first law of thermodynamics and mass conservation, the rate of pressure rise for constant mixture volume can be calculated with equation (3-16), according to the energy derivative of the mixture (Φ), the volume of the mixture (V_{mix}), the overall heat transfer (Q), any additional heat input (Q_w), the mass flow rate withdrawn from the tank (m_{out}), the latent heat of vaporization of the mixture (h_{lg}), the quality of the mixture withdrawn from the tank (x_{out}) and the densities of the liquid (ρ_{LH2}) and gaseous (ρ_{GH2}) phases [85].

$$\frac{dP}{dt} = \frac{\Phi}{V} \left[Q + Q_w - m_{out} \cdot h_{lg} \left(x_{out} + \frac{\rho_{GH2}}{\rho_{LH2} - \rho_{GH2}} \right) \right] \quad (3-16)$$

The energy derivative describes the change in internal energy under constant density, eq. (3-17). Figure 3-8 shows the evolution of Φ against mixture density for different pressure levels.

$$\Phi = \left[\rho_{mix} \left(\frac{\partial u}{\partial p} \right)_\rho \right]^{-1} \quad (3-17)$$

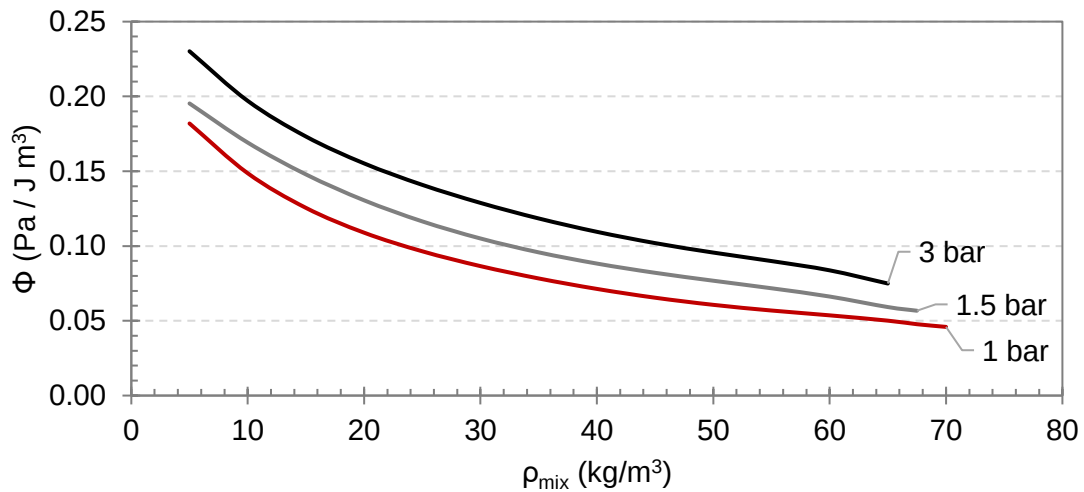


Figure 3-8 Energy derivative of an LH₂ and GH₂ mixture against mixture density at different pressure levels

Mixture properties effect on pressure change and pressure control mechanisms:

By breaking down the terms affecting the pressure rise, the following conclusions can be made:

- V:** The bigger the volume the lower the rate of pressure change. It is therefore preferable for the same quantity of fuel stored to use a low number of larger tanks than several smaller.
- Φ:** The lower the energy derivative the lower the rate of pressure change. Considering the Φ evolution from Figure 3-8, this can be achieved when the tank is operating at low LH₂ pressure and high mixture density. Correlating this with the aircraft application, at the start of the flight mission, where the tank is almost full of LH₂ (high ρ_{mix} and low pressure), the rate of pressure change can be as little as a quarter of that at the end of the mission where the tank is almost empty of LH₂ and full with GH₂ (low ρ_{mix} and pressure accumulation due to heat

leakage throughout the mission).

Q: Directly proportional, the higher the heat leakage the higher the pressure rise.

m_{out} : The higher the mass flow extracted from the tank, the lower the rate of pressure increase. Throughout a flight, the highest fuel flow demand occurs at take-off and slowly drops during the climb. At these phases, a pressure drop can be expected.

x_{out} : The quality of the withdrawn mixture is 0 if only LH_2 and 1 if only GH_2 is extracted. Extracting GH_2 from the tank would have much greater effect on pressure drop than simply extracting LH_2 . This technique has been widely used in LH_2 storage applications.

p_{LH_2} : The density change has a minor effect on the rate of pressure change.

Q_w : This represents the additional heat input in the mixture whether it is in the form of fuel mixing or heating. These two mechanisms can work as fail-safe devices that avoid pressure drops due to limited Q (overengineered insulation) or excessive withdrawal of tank contents.

3.3.3 Multipoint performance

The multipoint performance is a novel approach that aims to tackle the way the maximum pressure is determined and considers additional mission level sizing requirements. For the maximum pressure specifically, Verstraete [88] used it as an input and showcased the effect on tank weight for different ground hold times. Huete et al. [89] used the term dormancy time, which is the period that a tank can safely allow for pressure increase without the need for pressure control and sized the tank accordingly. However, both approaches tackled this only for a set of conditions, i.e. tank almost full with LH_2 and constant ambient temperature and pressure. In reality, these conditions change constantly throughout a flight and this effect needs to be taken into account when assessing the tank performance. These conditions are:

1. M_{mix} , since fuel is consumed in the engines

2. m_{out} , fuel flow demand
3. P_{amb} and T_{amb} , affecting the ΔP and the heat transfer respectively
4. P_{fuel} and T_{fuel} , due to pressure build-up

For the multipoint performance, the target time period is subdivided into several segments and then the heat transfer is estimated for each segment. For a sized tank and defined ambient conditions, the state of the tank contents at any given time (i) can be calculated according to equations (3-18) to (3-21) from each previous segment (i-1). For every operating condition, the volume of the mixture is constant and equal to the sized internal tank volume, eq. (3-18). The pressure of the mixture is calculated by adding the rate of pressure rise multiplied by the duration of the previous segment and the mixture pressure of the previous segment, eq. (3-19). The mass of the mixture is equal to the mass of the mixture of the previous segment less the mixture mass withdrawn from the tank, equation (3-20). Finally, the amount of evaporated LH₂ can be estimated based on the previous and current states of the LH₂ as well as the LH₂ withdrawn from the tank, as shown in equation (3-21).

$$V_{mix,i} = \text{constant} \quad (3-18)$$

$$P_{mix,i} = P_{mix,i-1} + \left(\frac{dP}{dt} \right)_i t_{seg,i-1} \quad (3-19)$$

$$M_{mix,i} = M_{mix,i-1} - (m_{engine} + m_{vent})_i t_{seg,i-1} \quad (3-20)$$

$$M_{evap,i} = M_{LH2,i-1} - M_{LH2,i} - m_{engine} t_{seg,i-1} \quad (3-21)$$

To summarise: two functions are developed for the tank sizing tool that has different input variables. The first sizes the tank, defines its geometry and weight and estimates the heat balance and pressure rise. Inputs that can be used are the wall and insulation properties, insulation thickness (t_{in}), any two of geometrical features or mass stored, fuel properties, and an indication of the mixture properties. Equivalently to gas turbine performance, this can be considered as a Design Point calculation.

The second uses the sized geometry as well as the state of the tank contents and then estimates the heat balance and pressure rise. The inputs, in this case, are the tank geometry and the mass and pressure of the tank contents as well as any hydrogen flow rates withdrawn from the tank. This is considered an off-design performance. Schematically the inputs are summarised in Figure 3-9.

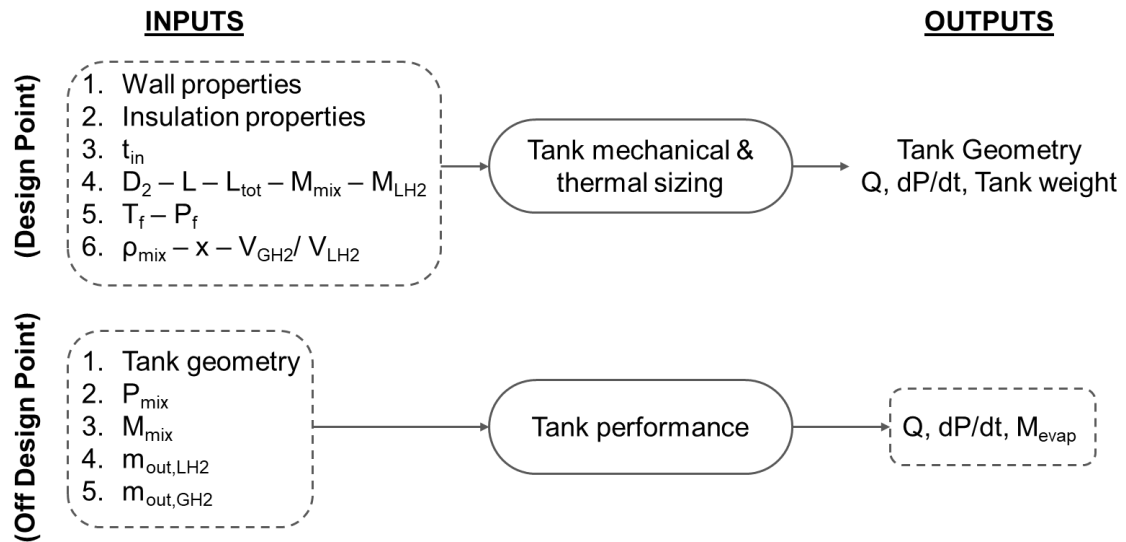


Figure 3-9 Schematic representation of the tank sizing and tank performance model

3.3.3.1 Model comparison with experimental data, accounting for temperature stratification

In this subsection, the model developed is compared against two experiments found in the literature involving the testing of stationary LH₂ tanks. The first experiment reported by Reynolds et al [118] featured a cylindrical tank and Styrofoam insulation. It had an internal volume of approximately 6.8 m³ and started with LH₂ occupying 15.3% of the internal volume. The tank was subjected to standard day ambient conditions and no hydrogen was withdrawn from the tank. Using the model described in the multi-point performance sequence, the pressure variation of the tank contents was predicted. In Figure 3-10 the experimental gauge pressure is shown together with the pressure predicted from the theoretical model. The predicted gauge pressure was however underestimated. Reynolds, using the heat transfer model described in [116] demonstrated similar levels of under-prediction.

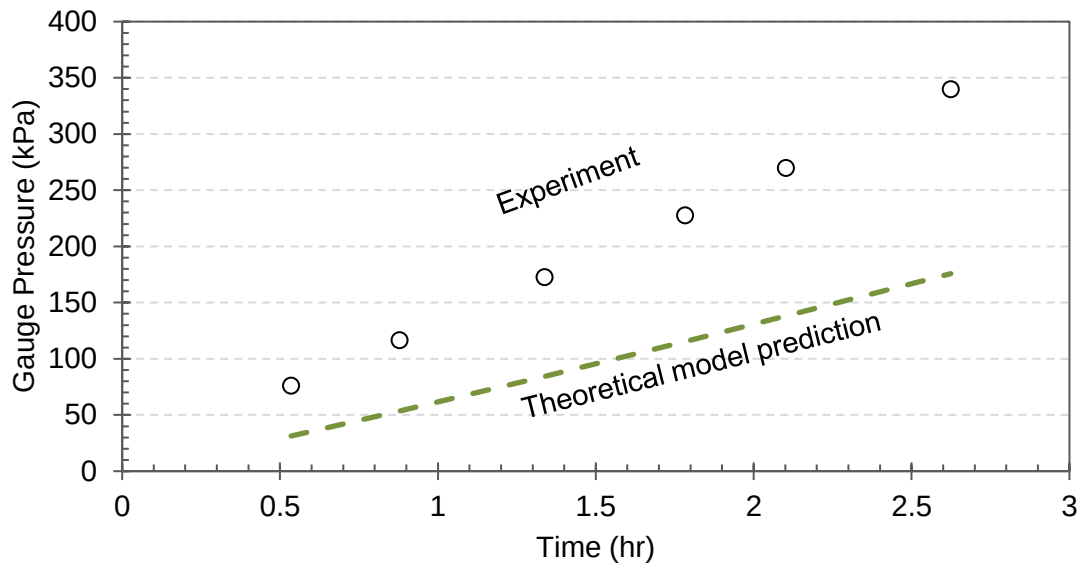


Figure 3-10 Model comparison with experimental data from [118]

In the second experiment, described by Hasan et al. [119], an ellipsoidal tank was subjected to ambient temperatures of 350, 280 and 80 K which corresponded to equivalent heat fluxes of 3.5, 2.0 and 0.35 W/m². For tests where no hydrogen was withdrawn from the tank, the LH₂ occupied approximately 84% of the total volume. By matching the reported parameters and conditions described in [119], the model predicted values that together with the corresponding experimental ones are presented in Figure 3-11 for the two higher heat flux cases. Again similar levels of under-prediction were encountered.

In both cases, the model was predicting the heat transfer based on the homogenous mixture assumption, however, this was not the case in the actual experiments, because temperature stratification occurred. In the ellipsoidal tank experiment [119], although the LH₂ showed little stratification from the bottom of the tank towards the fill level, about 2 K, the GH₂ temperature varied significantly, with a difference of 25 K from the fill level towards the top of the tank. As a result, the properties of the two phases differed depending on the internal vertical position in the tank, allowing a greater pressure rise than calculated for the homogenous state.

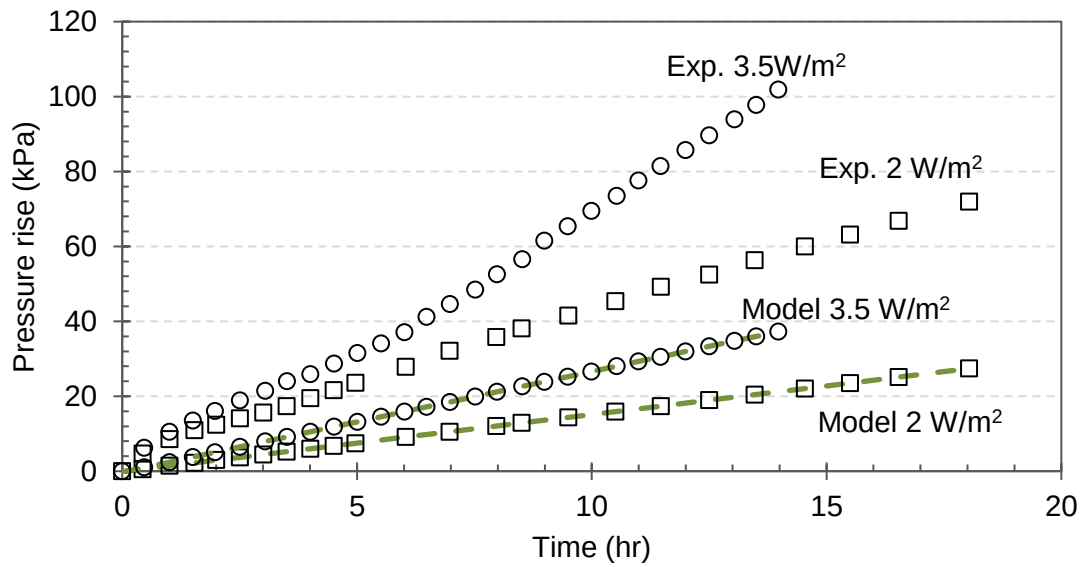


Figure 3-11 Model comparison with experimental data from [119] for heat fluxes of 3.5 and 2 W/m² for 350 K and 280 K ambient temperatures respectively.

3.3.3.2 Accounting for temperature stratification and mixing

In order to tackle the temperature stratification effect, Lin et al. [85] suggested multiplying the theoretical pressure rise, calculated in eq. (3-16), by a factor of 2, which was later also adopted in [99][26]. In the current study, however, an average stratification heat flux factor of 2.75 will be applied, as derived from the model comparison with the experiments shown in Figure 3-10 and Figure 3-11. This approach is expedient as it avoids having to model the separate temperatures of the liquid and gas in the tank.

Temperature stratification can be eliminated by mixing the tank contents. Reynolds, in his experiment [118], demonstrated this by leaving the tank for a while to warm up, then shaking it externally and then repeating the process a few more times. This effect is shown in Figure 3-12, where the results of the experiment are reproduced. After every mixing, the pressure of the hydrogen mixture dropped and matched that of the homogenous state predicted by the heat transfer model. Lin [85] also proposed either continuous or repeated mixing as a way to control the pressure rise, and estimated the pressure as in equation (3-16) by including the mixing power in the W term of the equation.

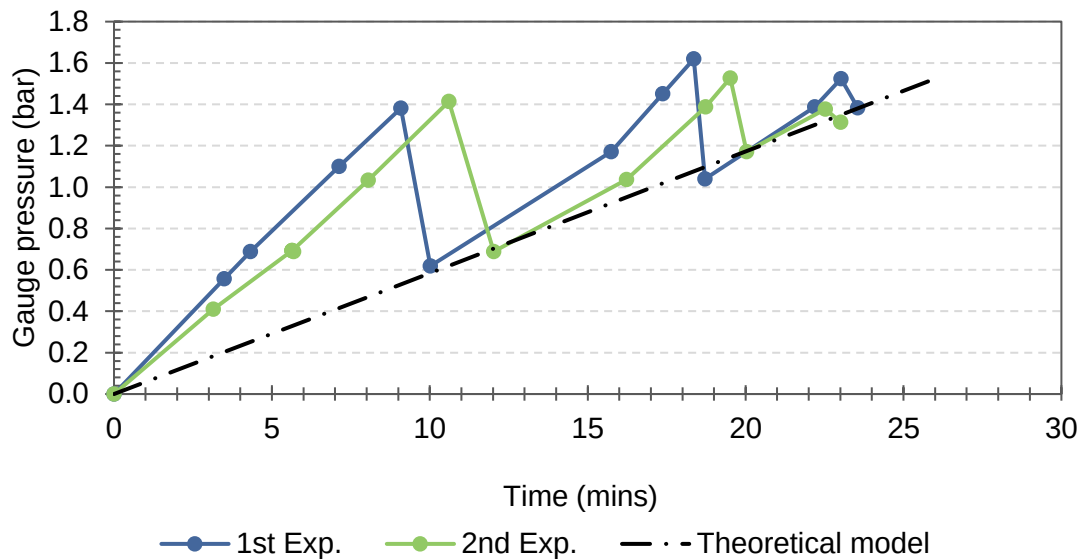


Figure 3-12 Reproduced from Reynolds 1957 [118], pressure variation of liquid nitrogen tank with external mixing against time

Considering aircraft applications, even in the case where LH₂ tanks are not equipped with internal mixing devices, there are several phases of a flight where forced external mixing can occur. For example, during accelerating or decelerating phases of the flight such as taxi, take-off, climb, descent and approach, sloshing of the tank contents could occur to produce mixing (even if baffles are used). Conversely, during overnight stays at the airport, or in ground hold or cruise phases of the flight, where the aircraft is immobile or flies at a steady speed, temperature stratification would be expected to occur inside the tanks.

Accounting for the mixing effect, the following process is adopted. First, the theoretical model pressure rise is estimated for all the examined segments. Then the heat transfer is calculated again from the beginning, only this time the stratification factor is included. If there is no mixing, the pressure of every segment is calculated from equation (3-16), but if mixing occurs, then the pressure is equal to the theoretical pressure estimated in the same segment. Schematically the process is shown in Figure 3-13.

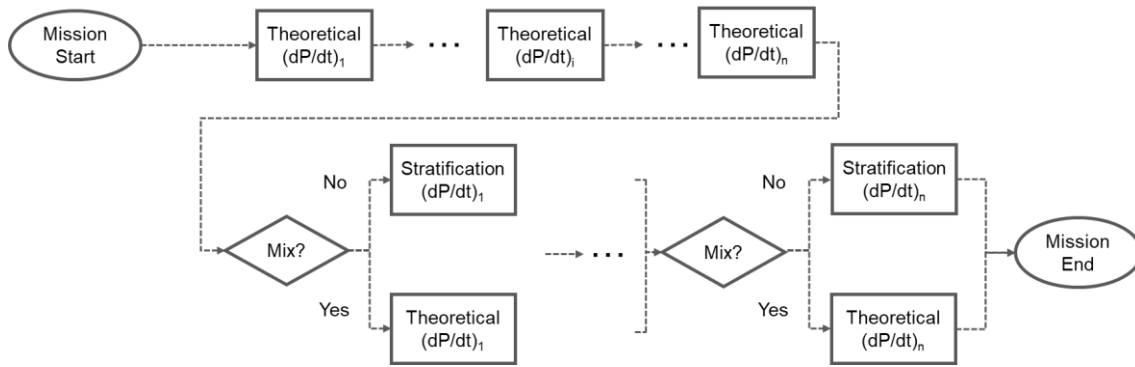


Figure 3-13 Workflow of tank pressure estimation throughout a mission

3.3.4 On flight application and optimisation

To demonstrate the tank sizing methodology and its optimisation a test case is presented. The tank is initially sized for a set of conditions before the flight and then based on the geometry the tank performance at mission level is performed. The tank needs to carry 5160 kg of LH₂ and the external diameter is set to 2.5 m. The diameter will remain a constant sizing input for the calculations that follow. Varying the diameter will impact the heat transfer as discussed in previous studies [88, 89, 96, 98, 99], however, although lighter tank designs can exist for a particular internal volume, their implementation is greatly restricted by their integration with the aircraft. Additional to the sizing conditions, a minimum ullage of 3% GH₂ by volume is also considered, as discussed by Brewer [20]. The initial filling pressure is set to 1.4 bar and is also the same for all the calculations. Ideally, a lower pressure would allow for higher density, lower volume and reduced differential pressure levels, but considering refuelling complications, 1.4 bar is considered to be a good compromise. Lastly, the maximum allowable pressure is assumed to be 300 kPa and the insulation is 8 cm of rigid polyvinylchloride closed-cell foam, with 0.0046 W/m² K thermal conductivity [96].

Table 3-6 Example tank sizing inputs

Parameter	Units	Value
M _{LH2}	kg	5160
M _{LH2,res}	kg	470

Parameter	Units	Value
$T_{amb,TO}$	$^{\circ}C$	30
D_2	m	2.5
V_{GH2} / V_{mix}	-	0.03
$P_{f,TO}$	bar	1.4
ΔP	bar	3
t_{in}	m	0.08

Additional considerations that affect the mission performance are also set:

- At the end of the mission, there needs to be 470 kg of LH_2 left for reserves.
- The engine fuel flow rate demand against the flight time is used as shown in Figure 3-14 normalised by its maximum value.
- Before take-off, a two-hour ground hold is considered with $30^{\circ}C$ ambient temperature.
- It is conservatively assumed that only the LH_2 would be usable by the engines.
- No GH_2 is vented throughout the flight.
- It is assumed that during taxi, take-off, climb, descent and landing, the tank contents are fully mixed.

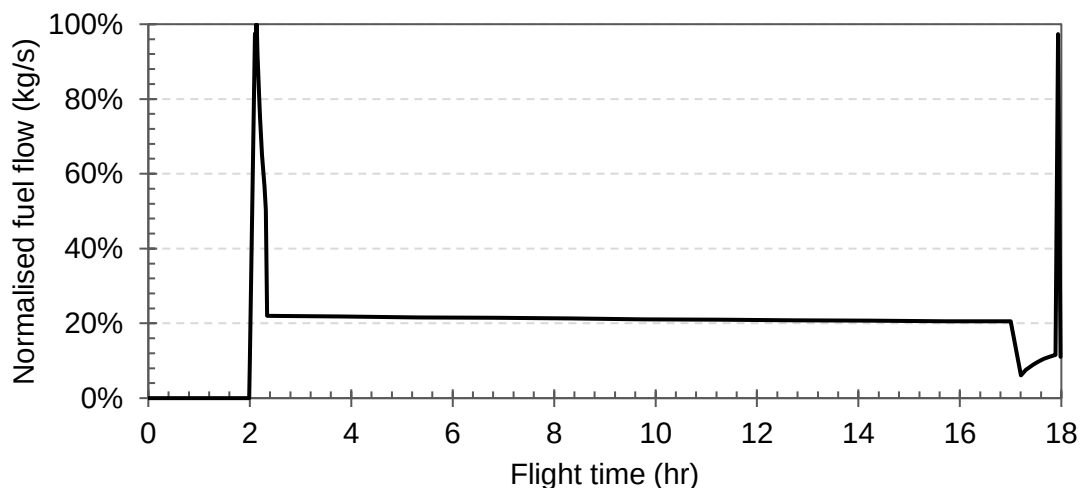


Figure 3-14 Normalised engine fuel flow demand of a mission profile

The flight mission is divided in several segments that will be used for the calculation, and the ambient temperature and pressure, as well as the LH₂ fuel flow demand per engine, are used as inputs.

3.3.4.1 Pressure evolution & wall optimisation

The sized tank has an overall length of 19.1 m and weighs 2.85 tonnes. Figure 3-15 shows the evolution of the absolute pressure of the hydrogen mixture versus time for three cases: the theoretical conditions, no-mixing conditions with the stratification factor discussed applied throughout the mission, and the case where intermittent mixing is taken into account.

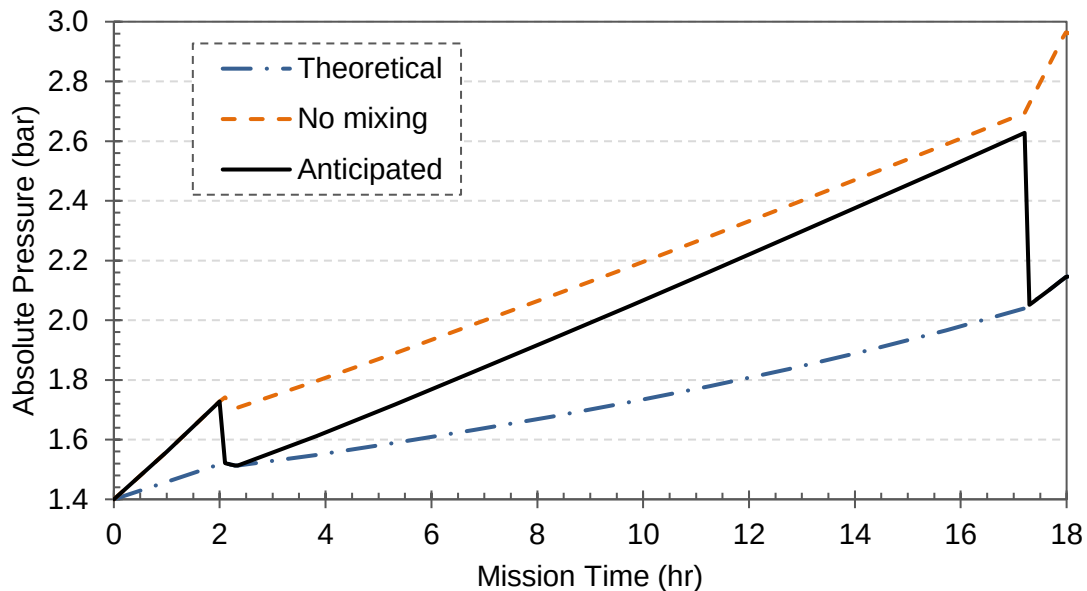


Figure 3-15 Absolute pressure evolution of hydrogen mixture.

More details about the mass and volume evolutions of the two phases, heat flux, energy derivative as well as pressure change throughout the examined mission are provided in Figure A-1 of the Appendix.

Key conclusions:

1. At ground hold, the rate of pressure rise is relatively high since no fuel is withdrawn from the tank and the ambient temperature is high.
2. During take-off and climb the rate of pressure rise is negative as high quantities of LH₂ are consumed by the engines leading to a small

absolute pressure drop for the theoretical and no-mixing cases. At the same time, the pressure drops even further due to mixing, to match the theoretically predicted value.

3. At cruise, there is an almost steady rate of pressure rise calculated, as the heat input to the tank is greater than the power extracted due to LH₂ fuel removal.
4. Finally, at descent and approach, a combination of increasing ambient temperature, low LH₂ fuel consumption and low LH₂ fill level leads to the highest rate of pressure rise across the whole mission, but only for a short amount of time. Also in the same segment, mixing provides a significant pressure drop.

Equally important as absolute pressure is consideration of the pressure difference between the fuel and the ambient air. Its evolution is shown in Figure 3-16 for the three mentioned cases. After take-off, the gauge pressure is increasing due to the decrease of ambient pressure with altitude, then it steadily increases at cruise due to the absolute fuel pressure increase, and finally, it drops with falling altitude through descent and landing.

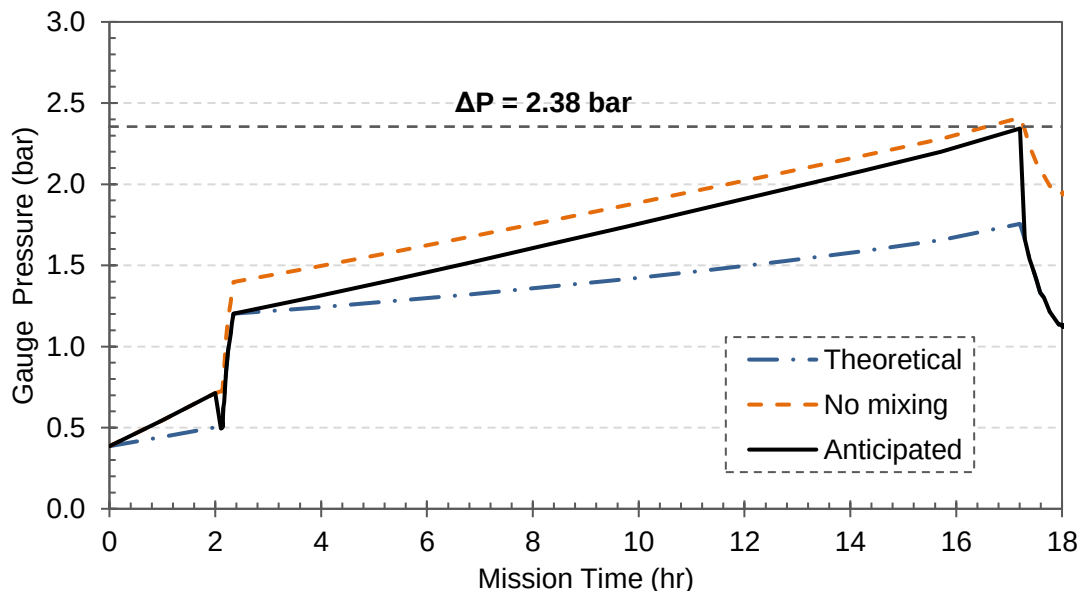


Figure 3-16 Gauge pressure evolution of hydrogen mixture.

From Figure 3-16 it is clear that the maximum pressure difference throughout the mission of 2.38 bar is met at end of cruise. Therefore there is no need to oversize size the tank wall for more than that value. In addition, by the end of the mission, 189 kg of LH₂ has been evaporated, leaving only 263 kg of LH₂, equivalent to 55% of the reserve fuel, inside the tank. To compensate for that a sizing loop is established with a twofold purpose:

1. Resizing the tank to compensate for the evaporated fuel
2. Sizing the wall thickness according to the maximum pressure difference estimated on the mission.

Figure 3-17 shows schematically the iterative process. The tank is first sized according to the useable LH₂ requirement. Then after calculating the tank performance for the mission, the remaining LH₂ is compared with the reserves and if it does not match, the calculation is repeated and at the same time, the wall is also resized according to the ΔP estimated from the previous iteration. The process repeats until the remaining LH₂ matches the reserve requirements and the optimum wall thickness for the insulation thickness and selected material is calculated.

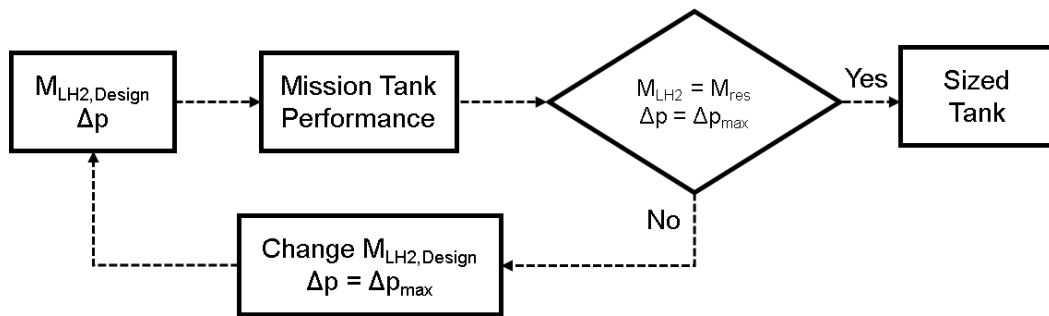


Figure 3-17 Tank sizing loop at mission level.

By applying this process, the tank now is lighter and weighs 2.45 tonnes, has an increased capacity of 5350 kg of LH₂ and therefore is longer with an overall length of 19.8 m.

3.3.4.2 Insulation thickness optimisation

So far it has been demonstrated how to minimise the wall thickness (and tank weight) to withstand the maximum pressure difference developed in the sizing

mission, and how the tank needs to be oversized to account for the evaporated LH₂, for a particular insulation thickness. To showcase how the insulation thickness affects the design, a parametric study is presented where the insulation thickness is varied between 5 and 25 cm. Figure 3-18 shows how the masses of the insulation, wall, evaporated LH₂ and the tank in total, vary with the insulation thickness.

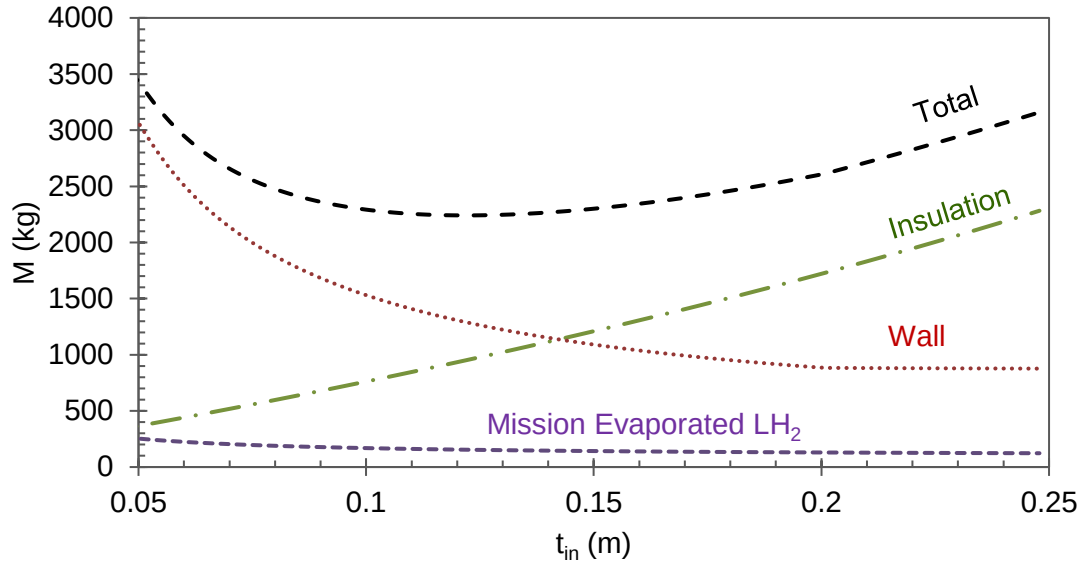


Figure 3-18 Wall, insulation, mission evaporated LH₂ and total tank weight of the tank against insulation thickness.

Starting with the insulation mass, as its thickness increases its mass increases as well. However, higher insulation thickness dictates lower heat transfer and therefore a lower maximum ΔP is developed, so lighter tank walls are possible. At the same time, lower heat transfer throughout the mission means less LH₂ is evaporated. This leads to an optimum point, in terms of weight, where for an insulation thickness of 12.4 cm the lightest tank design can be achieved, weighing 2.24 tonnes.

Another way to identify the best design is through the gravimetric efficiency (η_{tank}) as defined by Winnefeld [99]. This is calculated as the ratio of the useful LH₂ carried divided by the total LH₂ stored plus the weight of the tank, as shown in eq. (3-22). Contrary to the definition used by previous studies [88, 89, 98],

were in both the nominator and denominator of the efficiency definition the useful fuel was used, the definition presented in eq. (3-22) accounts for the unusable fuel that has been evaporated throughout a given period but remains as GH₂ inside the tank.

$$\eta_{\text{tank}} = \frac{M_{\text{usefull}}}{M_{\text{LH}_2} + M_{\text{total}}} \quad (3-22)$$

To select the best insulation material to use, four candidates proposed in [96] are examined. The same parametric study is performed, as before, using their typical properties, and the results are summarised in terms of tank efficiency in Figure A-2. Polyvinylchloride closed-cell foam offers the best solution, as it is at least 37% lighter than the other optimal designs and, due to lower insulation thickness, it also occupies the least space. It is also acknowledged that, although the gravimetric efficiency is affected by design considerations, such as tank dimensions (external diameter), filling pressure and ground hold time, varying them would still provide similar trends for the same insulation materials, and would probably dictate different optimum insulation thickness for each material.

3.3.4.3 Gaseous hydrogen extraction for pressure relief

In this section, the effect of GH₂ venting on the tank design is demonstrated. For the study that will follow, it is assumed that during cruise a certain quantity of GH₂ is vented overboard to restrict the pressure rise.

For the design discussed that led to the pressure evolution shown in Figure 3-16, the maximum pressure difference now is 123 kPa, as shown in Figure 3-19. This leads to a significantly lighter design (by 48%), but 38% more LH₂ is evaporated.

A parametric study is performed using polyvinylchloride insulation for the same thickness range as before. In Figure 3-20 the tank efficiency is plotted against insulation thickness with and without GH₂ venting at cruise and tank design parameters for the optimal designs are summarised in Table 2. The venting case provides lighter designs for low insulation thickness. Comparing the two

optima, the maximum allowable pressure is 22.4% if venting during cruise is enabled, which leads to the design of a 33.6% lighter tank. With GH₂ venting, although more fuel is evaporated throughout the mission, and therefore the tank needs to be sized to carry more LH₂ in the first place, lower insulation thickness is required which leads to a design that occupies 7.5% less external space. To avoid the pressure increase, 169 kg of GH₂ needs to be vented from the tank. This is equivalent to a potential 380 kW of thermal power per tank for the whole cruise segment. The GH₂ could be used in the engines or by auxiliary power units to meet aircraft secondary power requirements. In the results section, it is demonstrated how using that GH₂ as part of the engine fuel flow demand allows for further reduction in weight and external volume.

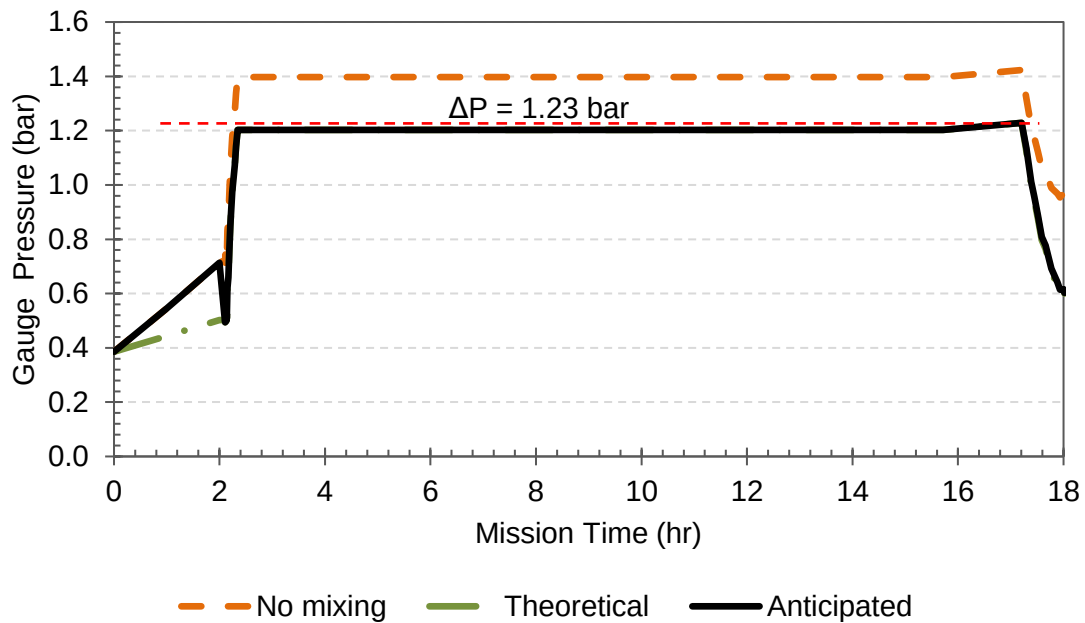


Figure 3-19 Gauge pressure evolution of hydrogen mixture, with cruise GH₂ venting.

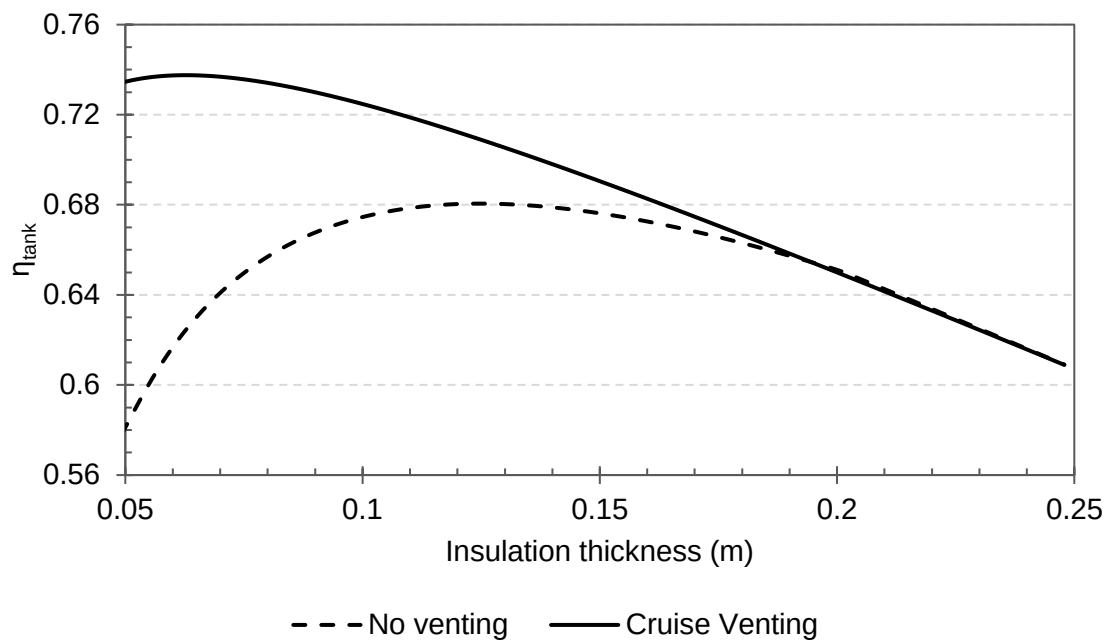


Figure 3-20 Tank gravimetric efficiency against insulation thickness, with and without cruise GH_2 venting, using polyvinylchloride foam.

Table 3-7 Tank sizing optimum results

	Units	No venting	Cruise venting	Change
Insulation thickness	(m)	0.124	0.063	-49.1 %
Maximum allowable pressure	(kPa)	162.4	126.1	-22.4 %
External volume	(m^3)	99.8	92.3	-7.5 %
LH_2 stored	(kg)	5313	5484	+3.2 %
Mission evaporated fuel	(kg)	153.4	323.5	+111 %
Mission vented fuel	(kg)	-	169.2	-
$M_{\text{evaporated}} / M_{\text{LH}_2, \text{stored}}$	-	3%	6%	
Total tank mass	(kg)	2242	1488	-33.6 %
η_{tank}	(-)	68%	74%	+8.4 %

3.4 Propulsion system sizing

This section includes a detailed description of the models for performance prediction and weight estimation used to size a TeDP configuration. It comprises six subsections:

1. BLI Fan performance estimation
2. Electrical component weight and efficiency estimation
3. Propulsion system subcomponents performance and weight estimation (gearbox and heat exchanger for fuel pre-heating)
4. TEDP mathematical model set up which integrates all of the above for performance prediction
5. Turbomachinery weight estimation
6. TEDP sizing framework that integrates all of the above for performance prediction and weight estimation

3.4.1 Boundary layer ingesting propulsor

In this section, the methodology and models to account for the performance of a boundary layer ingesting fan with variable area nozzle (VAN) installed over a flat surface are presented. The performance of such a propulsor in RPOOSIS (or any other OD propulsion system simulation software that is based on the same principles) can be modelled by three components:

1. BLI inlet module, to estimate the averaged flow properties at the inlet of the fan.
2. Fan module, to estimate the fan performance-based pre-existing BETA maps, as discussed by [104, 120].
3. Variable area nozzle (VAN) module, to allow for nozzle area change based on fan optimum performance.

The three components are connected through flow ports where information about the total pressure, temperature and mass flow is exchanged. Schematically the propulsor assembly is shown in Figure 3-21. The fan component is adopted from pre-existing libraries of PROOSIS (see TURBO library [120]). The BLI intake component was created in this context and its

mathematical model is detailed in 3.4.1.1. The VAN component is based on an existing module of PROOSIS but was modified to account for the variable nozzle area effect, which is described in 3.4.1.2.

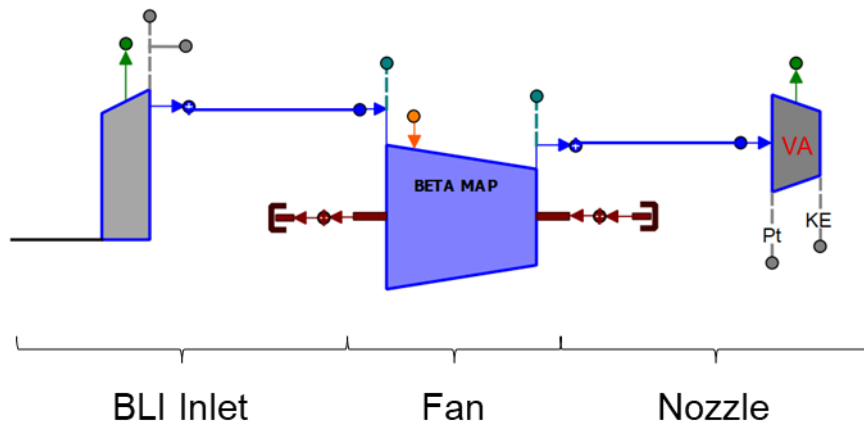


Figure 3-21 Remote BLI fan module as created in PROOSIS

3.4.1.1 BL intake component

The main purpose of the BL intake component is to estimate the key parameters that affect the fan performance, namely the mean total pressure, temperature and velocity at the fan face. The methodology used for the accounting is adapted directly from Goldberg [27, 78, 79] as its low fidelity approach makes it ideal for design space exploration where the fan size is expected to constantly change. Although boundary layer propulsive configurations are conceptualised to be highly integrated with the airframe, in the context of this approach they are separated. The airframe drag and thrust requirements are estimated based on a “clean” aircraft without any propulsive devices and the drag reduction and propulsion system benefits, as well as integration penalties, are accounted for in the propulsion system. To achieve that, the control volume of a BLI propulsor, shown in Figure 3-22, is considered. The station numbering represents the following:

- 0: Local ambient conditions with no interference with the airframe
- i: Position where the fan operation does not affect the development of boundary layer, distance is assumed to be 2x the fan diameter [79]

- 1: Fan intake inlet
- 2: Fan face
- 3: Fan exit
- 9: Nozzle exhaust

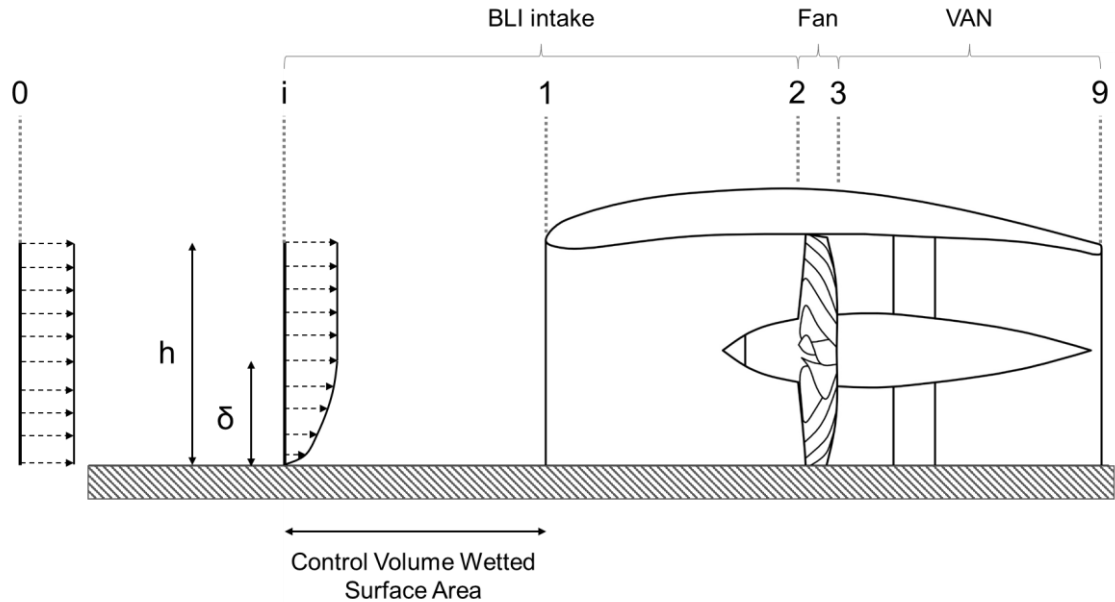


Figure 3-22 Boundary layer ingesting propulsor control volume

To express the thrust produced by the propulsor unit the term net propulsive force (NPF) is used. It comprises the gross thrust produced at the fan nozzle exit (FG_9), the momentum drag due to the mass flow at the inlet of the propulsor (FD_i), the skin friction drag between station i and 1 of the control volume ($\tau_{w,i}S_{w,i}$), the nacelle drag (FD_{nac}) and the skin friction drag of the whole control volume (ΔFD). The latter term seemingly adds to the NPF but effectively is used to avoid double accounting for the skin friction drag already considered in the airframe drag prediction.

$$NPF = FG_9 - FD_i - \tau_{w,i}S_{w,i} - FD_{nacelle} + \Delta FD \quad (3-23)$$

The terms FG_9 and FD_i are depended on the fan performance and the inlet local conditions and the terms $\tau_{w,i}S_{w,i}$, FD_{nac} and ΔFD on the ambient and local conditions as well as the sizing of the propulsor. The BLI inlet module, apart

from the flow properties, needs to also estimate the terms FD_i , $\tau_{w,i}S_{w,i}$, FD_{nac} and ΔFD of eq. (3-23).

BL control volume:

The BLI inlet module utilises of the following process to determine the control volume and estimate the flow properties for a given mass flow requirement, m_{total} .

The boundary layer thickness as developed over a flat surface is dependent on the local ambient conditions (i.e. the Reynolds number) and the distance of interest from the beginning of the flat plate (l_{plate}). An approximation to the boundary layer thickness for a simple turbulent flat plate model is adopted from [121] and is calculated by eq. (3-24), where k is a correction factor. According to Stratford et al. [121] k is 0.23, however, Goldberg corrected this value to fit the CFD data of the N3-X [122] based on the local free stream Mach number (M_{local}).

$$\delta = \frac{kl_{plate}}{Re^{1/6}}, \text{ for } Re > 10^7 \quad (3-24)$$

The velocity profile is approximated using the 1/n law and as such the velocity (V_y), density (ρ_y) and total pressure (P_y) at any given height (y) from the surface are estimated based on equations (3-25) to (3-27) respectively.

$$\frac{V_y}{V_0} = \left(\frac{y}{\delta} \right)^{1/n} \quad (3-25)$$

$$\frac{\rho_y}{\rho_0} = \left[1 + \frac{\gamma-1}{2} M_0^2 \left[1 - \left(\frac{y}{\delta} \right)^{2/n} \right] \right]^{-1} \quad (3-26)$$

$$\frac{P_y}{P_0} = \left[1 + \frac{\gamma-1}{2} M_0^2 \left[1 - \left(\frac{y}{\delta} \right)^{2/n} \right] \right]^{\frac{-\gamma}{(\gamma-1)}} \quad (3-27)$$

The mass flow at any y can then be determined by eq. (3-28), where w_{BL} is the control volume width. To specify the control volume, Goldberg et al. [78] assumed that the boundary layer height at the design point is the same as the fan diameter and calculated the width based on the mass flow equation. For off-design calculations, the width remained constant and the height varied. Felder et al. [122] on the other hand set the width of the control volume to be equal to the fan diameter and reverse calculated the height. In this context, the latter approach is followed, however, it is acknowledged that a better definition of the width of the control volume is only possible through higher fidelity 3-D approaches.

$$m_y = \rho_0 V_0 w_{BL} \int_0^y \frac{\rho_y}{\rho_0} \frac{V_y}{V_0} d(y) \quad \text{or} \quad m_{BL} = \rho_0 V_0 w_{BL} \int_0^\delta \frac{\rho_y}{\rho_0} \frac{V_y}{V_0} d(y) \quad (3-28)$$

By setting y to δ , the boundary layer mass flow, m_{BL} , can be estimated. If m_{BL} is bigger than the mass flow requirement, then (3-28) is iterated until a y lower than δ is found that allows for m_y to be equal m_{total} . If m_{BL} is lower than m_{total} then free stream air must also be considered in the control volume and the term mass flow group (MAG) can be used to assist the calculations. It represents the mass flow ratio of viscous to inviscid flow of the boundary layer, eq. (3-29). By combining (3-28) and (3-29), the h can be calculated as in eq. (3-30).

$$MAG = \frac{m_{BL}}{\rho_0 V_0 \delta w_{BL}} \quad (3-29)$$

$$\frac{h}{\delta} = \frac{m_{total}}{\rho_0 V_0 \delta w_{BL}} - MAG + 1 \quad (3-30)$$

Once the control volume height has been determined, the mean pressure and velocity averaged for mass flow can be estimated and at any given y these are calculated by eq. (3-31) and (3-32) respectively.

$$\frac{\bar{P}_y}{P_0} = \frac{1}{m_y} \int_0^y \frac{P}{P_0} dm \quad \text{or} \quad \frac{\bar{P}_y}{P_0} = \frac{\rho_0 V_0 \delta w_{BL}}{m_y} \int_0^{y/\delta} \frac{P_y}{P_0} \frac{\rho_y}{\rho_0} \frac{V_y}{V_0} d\left(\frac{y}{\delta}\right) \quad (3-31)$$

$$\frac{\bar{V}_y}{V_0} = \frac{1}{m_y} \int_0^y \frac{V_y}{V_0} dm \quad \text{or} \quad \frac{\bar{V}_y}{V_0} = \frac{\rho_0 V_0 \delta w_{BL}}{m_y} \int_0^{y/\delta} \frac{\rho_y}{\rho_0} \left(\frac{V_y}{V_0} \right)^2 d \left(\frac{y}{\delta} \right) \quad (3-32)$$

Similarly to the mass flow if $h > \delta$, then eq. (3-33) and (3-34) are used where P_{BL} and V_{BL} are determined according to (3-31) and (3-32) respectively by setting $y = \delta$.

$$\frac{\bar{P}_i}{P_0} = \frac{\frac{h}{\delta} - 1 + \frac{\bar{P}_{BL}}{P_0} MAG}{\frac{h}{\delta} - 1 + MAG} \quad (3-33)$$

$$\frac{\bar{V}_i}{V_0} = \frac{\frac{h}{\delta} - 1 + \frac{\bar{V}_{BL}}{V_0} MAG}{\frac{h}{\delta} - 1 + MAG} \quad (3-34)$$

Once the control volume has been defined the force terms of eq. (3-23) apart from the gross thrust can be calculated. Momentum drag (FD_i) depends on the mass flow, the mean velocity the area of the nozzle as well as the static pressure difference of the flow and the ambient condition, eq. (3-35). Compared to a non-BLI configuration, the momentum drag is reduced by the ratio between the mean velocity at station i and the free stream velocity. The skin friction drag of the control volume is a function of the local skin friction coefficient (C_f), the free-stream properties and the surface area of consideration ($w_{BLD_{fan}}$), eq. (3-36). The nacelle drag is estimated according to the nacelle skin friction coefficient ($C_{f,nac}$) and the wetted surface area of the nacelle ($w_{nac|nac}$ if a flat surface is considered, or $\pi w_{nac|nac}$ if axi-symmetric). The ΔFD term is calculated as shown in eq. (3-38), based on the skin friction coefficient of the fuselage, the free stream properties and the surface area covered by the propulsor and its control volume.

$$FD_i = m_{total} \times \bar{V}_i \quad (3-35)$$

$$\tau_{w,i} \times S_{wet} = C_f \frac{1}{2} \rho_0 V_0^2 \times w_{BL} D_{fan} \quad (3-36)$$

$$FD_{nac} = C_{f,nac} \frac{1}{2} \rho_0 V_0^2 \times (w_{nac} l_{nac}) \quad (3-37)$$

$$\Delta FD = C_{f,fus} \frac{1}{2} \rho_0 V_0^2 \times (w_{BL} D_{fan} + w_{nac} l_{nac}) \quad (3-38)$$

It is noted that the total pressure and velocity change between stations i and 2, ambient static pressure variation in the vicinity of the exhaust (affecting the calculation of the gross thrust, FG_9) as well as penalties in fan efficiency due to BL interaction are not addressed in the context of this methodology. Instead a constant deficit in pressure and fan efficiency may be included based on detailed CFD and experimental applications found reported in literature.

Design point and off-design calculation of the control volume:

For off-design calculations, where the fan size is established, equations (3-24) to (3-34) can be used, since the width, w , is determined. The height of the control volume then varies until the mass flow requirements are met. At design point, however, since the fan size is calculated, some extra steps are considered. Assuming that the mass flow capture ratio is 1, the ratio of areas between station 1 and i is given by eq. (3-39), where M_2 is the Mach number at the fan face and is a design consideration. Then the fan diameter (and control volume width) is calculated by eq. (3-40) based on the hub to tip ratio, (HTR), of the fan.

$$\frac{A_2}{A_i} = \frac{M_i}{M_2} \left(\frac{1 + \frac{\gamma-1}{2} M_2^2}{1 + \frac{\gamma-1}{2} M_i^2} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \quad (3-39)$$

$$w_{BL} = D_{fan} = \sqrt{\frac{4A_2}{\pi(1-HTR^2)}} \quad (3-40)$$

According to the value of w_{BL} , the properties of the control volume can be recalculated. The whole process is then iterated several times until the calculated w_{BL} does not significantly vary from the previous iteration. Schematically the design point and off-design calculation are presented in Figure 3-23.

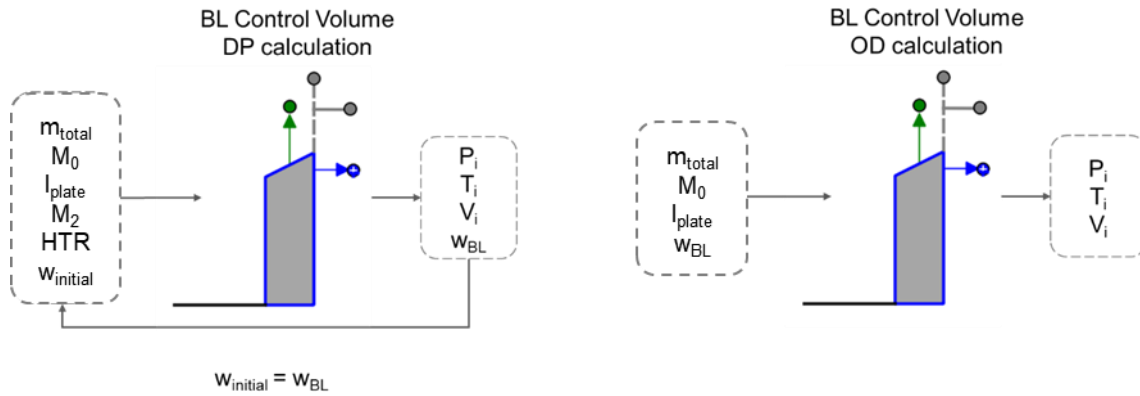


Figure 3-23 Schematic representation of BL control volume calculation at design and off-design points

3.4.1.2 Variable area nozzle (VAN)

In this section the methodology followed to account for the variable nozzle area is presented. In a distributed propulsion configuration moderating the rotational speed of a remote propulsor has significant importance as allows for the smooth operation of the BLI fan by avoiding surging and optimising fan performance.

As briefly The N3-X and subsequent based studies suggested that the rotational speed can be controlled with the use of inverters. This electrical equipment allows the modification of the current frequency and as a result the mechanical speed of the coupled fan. However, such devices add significant weight and penalise the transmission efficiency. Apart from the transmission system, the fan operation can be moderated by utilising variable pitch geometry on the fan or by varying the exhaust nozzle area or with a combination of these two. This study examines the latter.

In a conventional turbofan with fixed nozzle geometry, the off-design operation can be determined by a single control variable. By varying the nozzle area, however, another control variable is added that needs to be determined for every calculation. In a distributed propulsion configuration if every remote fan utilises VAN, then an equivalent number of control variables is added. Considering that every remote fan could potentially operate under slightly different conditions then each nozzle area would require a different input, which significantly increases the complexity of the calculation. To avoid that, a methodology is introduced that allows for an automated nozzle area selection that maximises the efficiency of the fan, for any operating conditions, and is generic so it can be applied for any fan sizing.

The nozzle area selection can be achieved via optimising each fan performance but adds both complexity and computational time to the off-design calculation. A faster and simpler approach of equivalent accuracy is the selection of the nozzle area (that corresponds to peak efficiency performance) based on a predefined map with a set of parameters as introduced by Chapman et al. [123] and subsequently Alexiou et al. [124].

To create such a map a two-step process is introduced:

Step 1: The propulsor module is sized based on a set of design point conditions and the BL width, the fan map scaling factors, as well as the nozzle area, are calculated. More details about the mathematical model of the design point sizing of the propulsor will be provided subsequently in section 3.4.4.1.

Step 2: The nozzle area is set as unknown variable and is calculated based on two varying parameters:

1. Set of speedline and corrected mass flow that correspond to the peak efficiency points of the baseline fan map found in [125], as shown in Figure 3-24. Contrary to a fixed nozzle area configuration the rotational speed is varied and the mass flow is then determined by the nozzle exit area. In this case, as the nozzle area needs to be determined, both the rotational speed and mass flow are provided.
2. Free stream Mach number. As the mass flow used as input in the first set is corrected based on inlet temperature and pressure, (or equivalently altitude) only the Mach number affects the nozzle area calculation [123, 124].

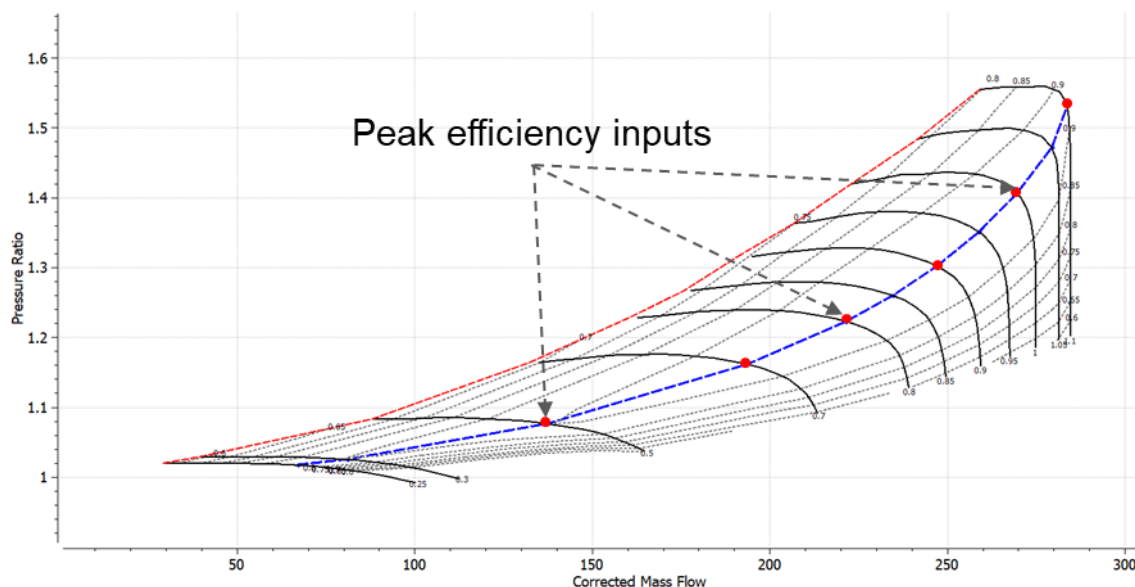


Figure 3-24 Remote fan peak efficiency operating line

As the nozzle areas for the different sets of conditions are calculated via the process described, the schedule can then be created based on two interpolating parameters. Naturally, these two can be the speedline of the fan and the free stream Mach number, which were used in the first place to map the nozzle areas (also used by [123, 124]). However, using speedline information, which is a fan component variable, in the nozzle component, will require changes to both the nozzle and fan component. Therefore, another interpolating parameter that is based on nozzle related information is chosen in this context.

Using the steps discussed, a BLI propulsor is at first sized based on the inputs presented in Table A-1 of the appendix. Then the nozzle area is calculated for different fan peak efficiency operating points and Mach numbers. Plot (a) of Figure 3-25 presents the evolution of the ratio of the nozzle area to its DP value (A_{OD} / A_{DP}) against the nozzle pressure ratio (NPR) and plot (b) of the same figure against the ratio of the nozzle corrected mass flow to its DP value ($W_{cor}/W_{cor@DP}$). For low Mach numbers, 0.1 – 0.2, as the power setting is increasing the nozzle area needs to increase at first and then close back down to its design conditions. This translates that for taxi and take-off conditions a bigger opening is required to maximise efficiency and thrust. At high Mach numbers, above 0.6, as the power setting is increasing, the nozzle exit area needs to increase as well but only to the point that it matches approximately its design value. High Mach numbers are only met during cruise and top of climb where the power settings are relatively high and therefore almost no change to the nozzle area is required. At intermediate Mach numbers, 0.3 – 0.5, the area varies. At high power settings, usually met during the climb segment, the area can be slightly bigger than its design value, but at low power settings, usually met during descent, the opposite applies.

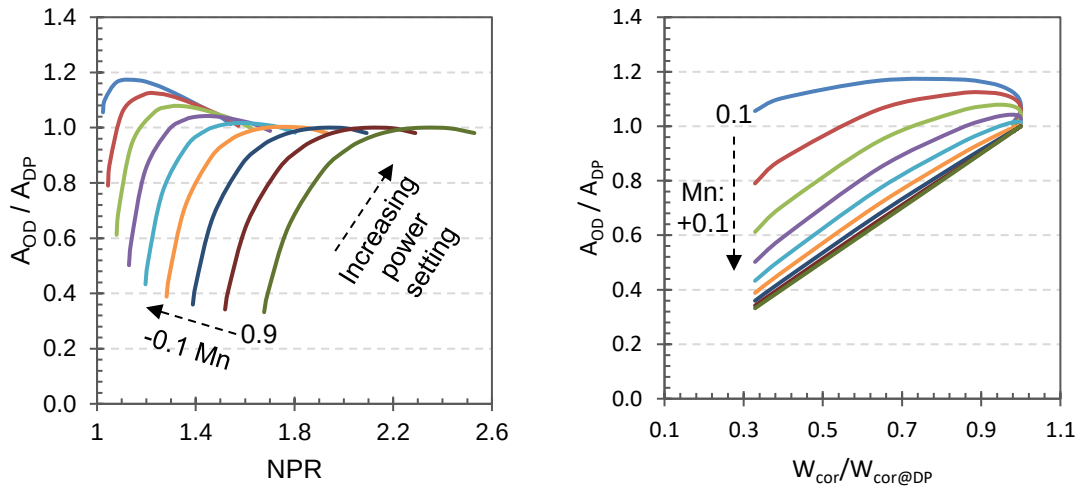


Figure 3-25 Nozzle area opening relative to DP against NPR and W_{cor} relative to its DP value for different Mn

Out of NPR and $W_{cor}/W_{cor@DP}$, the latter is preferred as it allows for better data tabulation.

To integrate this into a propulsion system simulation software the following equations, (3-41) – (3-43), need to be included in the nozzle component:

$$W_{cor_ratio} = \frac{W_{cor}}{W_{cor@DP}} \quad (3-41)$$

$$A_{ratio} = F(Mach, W_{cor_ratio}) \quad (3-42)$$

$$A_{OD} = A_{DP} \cdot A_{ratio} \quad (3-43)$$

For a design point calculation, $W_{cor@DP}$ and A_{des} are calculated and W_{cor_ratio} as well as A_{ratio} are set to 1. At off-design calculations, $W_{cor@DP}$ and A_{DP} are provided by the design point and W_{cor_ratio} as well as A_{ratio} are calculated. For the A_{ratio} mapping, the discussed range of fan power settings and Mach numbers needs to be calculated after the sizing.

3.4.2 Superconducting Electrical Components

For the electrical motors and generators, two areas are considered that contribute: the sizing and weight estimation to account for the added weight, and the efficiency prediction which affects both the engine sizing and its performance throughout the flight envelope.

Among the different types of electrical machine architectures (e.g. asynchronous, switched reluctance or synchronous, for aircraft applications a plethora of previous studies [64, 66, 67, 126, 127] selected synchronous machines. With regards the superconductivity, a synchronous machine can be partly or fully superconducting. The former applies when the windings of only the rotor or stator comprise superconducting materials, as proposed by Vratny et al. [126], and the latter when both do, as discussed by Brown [67]. In this context, and similar to [126], permanent magnet synchronous machines are chosen (superconducting windings on the stator and permanent magnets on the rotor).

3.4.2.1 Sizing and weight estimation

The sizing requirements of a motor or a generator are the maximum required power and a maximum rotational speed. Although maximum power may not coincide with maximum rotational speed, in the context of aircraft applications, these measures are used for the sizing. The power of the shaft (PW_{out}) is a function of the torque (Q_R) and the rotational speed (ω), eq. (3-44) and the torque in turn is a function of shear stress (force density over rotor surface area) (σ_R) and the volume of the rotor (V_R), eq. (3-45). Finally the shear stress is defined by the electrical loading (K_s), the peak of fundamental magnetic flux waveform at the airgap (B_g) produced by the rotor and the winding factor (k_l), eq. (3-46).

$$PW_{out} = Q_R \omega \quad (3-44)$$

$$Q_R = 2\sigma_m V_R \quad (3-45)$$

$$\sigma_R = \frac{k_l K_s B_g}{\sqrt{2}} \quad (3-46)$$

By combining equations (3-44) to (3-46) the volume of the rotor is expressed as in eq. (3-47) [60]. Therefore, for a given power requirement, a compact and lighter machine can be designed by a combination of increased electrical loading, magnetic flux and rotational speed. The typical winding factor for a high-efficiency machine is considered to be 0.93 [128].

$$V_R = \frac{PW_{out}}{\sqrt{2}k_l K_s B_g \omega} \quad (3-47)$$

The electrical loading is estimated according to the geometry of the teeth and slots of the stator (packing factor (k_c), depth (d_{slot}) and width (w_{slot}) of the slot as well as tooth width (w_{tooth})) and the current density of the conductor (J), eq. (3-48). The conductor packing factor, eq. (3-52), represents how efficiently the conductor is stacked in a slot, with typical values varying between 0.3 – 0.8 [128]. A cross-section of the stator together with a schematic representation of a slot and a tooth is shown in Figure 3-26.

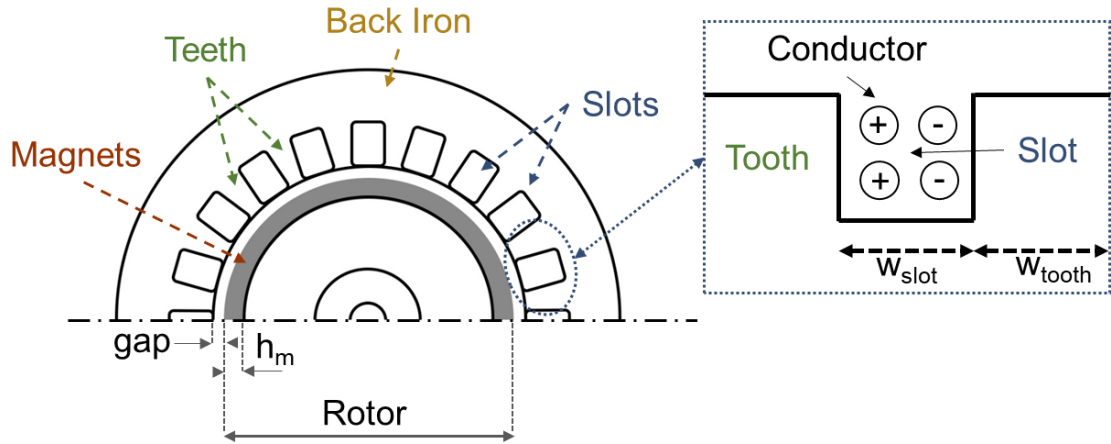


Figure 3-26 Cross-section of a permanent magnet machine and slot schematic

$$K_s = d_{slot} J k_c \frac{w_{slot}}{w_{slot} + w_{tooth}} \quad (3-48)$$

$$k_c = \frac{A_{conductor}}{A_{slot}} \quad (3-49)$$

The electrical loading can be maximised by optimising the slot and tooth geometry. In more detail, increasing slot depth can have a desired increase in the electrical loading but can also lead to wider and heavier stator geometries. Another approach would be to maximise the slot to tooth width to a point where the stator can be designed with no teeth in a slotless configuration [60, 129].

Additionally the electrical loading also depends on the current density of the conductor. Typical current densities for conventional copper conductors vary between 5 – 30 A/mm² depending on the cooling applied [60]. Superconductors however can achieve much higher values when operating at cryogenic temperatures. In the context of this study magnesium diboride (MgB₂) is considered since its limits of operating temperature and therefore cooling needs are met by the LH₂ stored conditions. As in [67, 130], the current density for MgB₂ can be calculated as in eq. (3-50) for an operating temperature (T) between 20 – 28 K and a peak of fundamental magnetic flux (B_g) that does not exceed 3T.

$$J = (1.907T^2 - 121.5T + 2027.9) - (1.0268T^2 - 64.414T + 116) \ln(B_g) \quad (3-50)$$

Finally, the magnetic field strength in the gap between the rotor and the stator for a permanent magnet machine is a function of the gap, the height of the magnet (h_m) and the remnant flux density of the magnet (B_r) with typical values between 0.8-1.3 T.

$$B_g = B_r \frac{h_m}{h_m + gap} \quad (3-51)$$

Overall equations (3-44) to (3-51) enable the estimation of V_R and based on the geometry of the cylinder the same volume can be achieved with a combination of rotor length and diameter. The diameter is restricted by the maximum mechanical stress caused by the centrifugal force, eq. (3-52), which depends yield strength of the material (σ_y), its density (ρ) rotational speed, a safety factor

(k_s) and a constant depended on the materials poison ratio, which for steel cylinder with a small bore is considered 0.823 [126, 128]. Additionally, the shaft diameter is also restricted by a maximum linear tip speed of 250 m/s [131]. The maximum diameter is then selected as the lower of the two maxima estimated by equations (3-52) and (3-53).

$$R_{\max,1} = \sqrt{\frac{\sigma_y}{k_s C' \rho \omega^2}} \quad (3-52)$$

$$V_{tip} = \omega_{\max} R_{\max,2} < 250 \text{ m/s} \quad (3-53)$$

The length of the shaft is restricted by its mechanical resonances and the critical angular speeds that these occur. The maximum length that can be considered that guarantees operation below the first critical speed is calculated by eq. (3-54) according to the maximum rotational speed, the modulus of elasticity (E), the density, the moment of inertia (I) and the cross-sectional area of the shaft (S) as well as a factor of safety (k_s) [128].

$$L_{R,\max} = \frac{\pi^2}{k_s \omega_{\max}} \sqrt{\frac{EI}{\rho S}} \quad (3-54)$$

Combining the three limitations any diameter can be selected that is bigger than the minimum associated with the maximum length and the maximum dictated by equations (3-52) and (3-53). Vratny et al. [126] demonstrated that between the two limits, choosing the bigger diameter results in slightly lighter designs overall, however, in the context of this work the minimum is selected as integration aspects are taken into account.

Finally, by combining the three equations and setting $R_{\max}$ equal to $R_{\min}$, for any given power requirement a maximum allowable rotational speed can be estimated. Any speed beyond that suggests an infeasible rotor design as the maximum diameter due to centrifugal forces is less than the minimum diameter associated with the maximum length.

The overall weight of the electrical machine can be estimated as sum of the individual components, eq. (3-55).

$$M_{total} = 1.5(M_{stator} + M_{rotor} + M_{magnets} + M_{conductor} + M_{services}) \quad (3-55)$$

- The weight of the rotor, M_{rotor} , is calculated based on its geometry determined by equations (3-44) - (3-54) with the material properties suggested in [126].
- The weight of stator, M_{stator} , is estimated according to the methodology outlined by [131, 132] and the material properties in [126].
- The weight of the magnets, $M_{magnets}$, and the conductor, $M_{conductor}$, is based on the methodology of [132].
- The services weight, $M_{services}$, accounts for the mountings and the bearings required and is assumed to be 13% of the combined aforementioned weights [133].
- A 50% of the total weight is also added to consider the weight of non-electromagnetic components [67].

3.4.2.2 Efficiency estimation

For the estimation of the efficiency of a permanent magnet high temperature superconducting machine, the power losses are broken down in the following categories:

- AC stator losses, PW_{AC} , associated with the conductor and are subdivided into three categories:
 - Hysteresis losses
 - Normal eddy currents
 - Coupling eddy currents
- Windage losses, PW_{wind}
- Stray losses, PW_{stray}
- Miscellaneous losses (bearing friction etc), PW_{misc}

The stator losses are calculated using the methodology outlined by Brown [67] for MgB_2 and primarily depend on the conductor geometry, current frequency, (f), magnetic field strength, (B), current density, (J). The current frequency can be calculated as in eq. (3-56) as a function of the number of pole pairs (p) and

the rotational speed. Windage and miscellaneous losses are scaled up linearly as in [126], based on a 1 MW motor found in [134]. Stray losses are also scaled in the same way but for an HTS machine a 50% reduction is considered [126]. The total losses, P_{loss} , are calculated as in eq. (3-57) and the efficiency in eq. (3-58).

$$f = \frac{p\omega_m}{2\pi} \quad (3-56)$$

$$PW_{loss} = PW_{AC} + PW_{Wind} + PW_{Stray} + PW_{Misc} \quad (3-57)$$

$$\eta_{HTS} = \frac{PW}{PW + PW_{loss}} \quad (3-58)$$

3.4.2.3 Design space exploration and mapping

In this section, an example of the discussed sizing methodology is presented to showcase the design space of HTS machines with a range of rated power between 1 and 50 MW as well as rotational speeds between 500 and 10000 rpm. The list of assumptions used is presented in Table 3-8. A 3-phase configuration is chosen as it could potentially lead to slightly lighter designs [131]. For the example alone, three pole pairs are selected, however, in the test cases that will follow the pole pairs are to be decided based on the best matching between motor and generator. The operating temperature is 25 K, which is considered a feasible value given the stored hydrogen properties. The packing factor (k_c) is chosen on the low side of the available range, since HTS armature of high-speed electrical machines needs to be well insulated. Regarding the rotor sizing and limitations, an overspeed of 50% is accounted that is both applied to the length and diameter limit together with a safety factor of 1.5. Finally, an additional length of 33% [126] as calculated in eq. (3-54) is considered to accommodate the mountings and bearings of the assembly.

Table 3-8 HTS sizing assumptions

Parameter	Symbol	Units	Value
Number of pole pairs	p	-	3
Winding factor [128, 132]	k_l	-	0.93

Parameter	Symbol	Units	Value
Operating temperature	T	K	25
Magnet remnant magnetic flux (see section A.3 of the Appendix)	B_r	T	1.30
Slot depth [132]	h_s	m	0.02
Packing factor	k_c	-	0.3
Tooth to tooth and slot width ratio	$w_t / (w_s + w_t)$	-	0.3
Airgap between rotor and stator [67]	gap	m	0.0015
Magnet height [131]	h_m	m	0.02
Safety factor	k_o	-	1.50
Rotor overspeed	-	-	50%

The results are presented in Figure 3-27 in terms of power to weight ratio (a) and efficiency (b) against rated power and rotational speed. The power to weight ratio increases with increasing power and rotational speed but at both high values, the design space is restricted by the mechanical limitations discussed. At the same time, the efficiency increases with increasing power and reducing rotational speed. The lower the rotational speed the lower the current frequency which in turn results in lower AC losses in the stator. Nevertheless, for the majority of the design space, the efficiency maintains a relatively high value attributed to the superconducting technology.

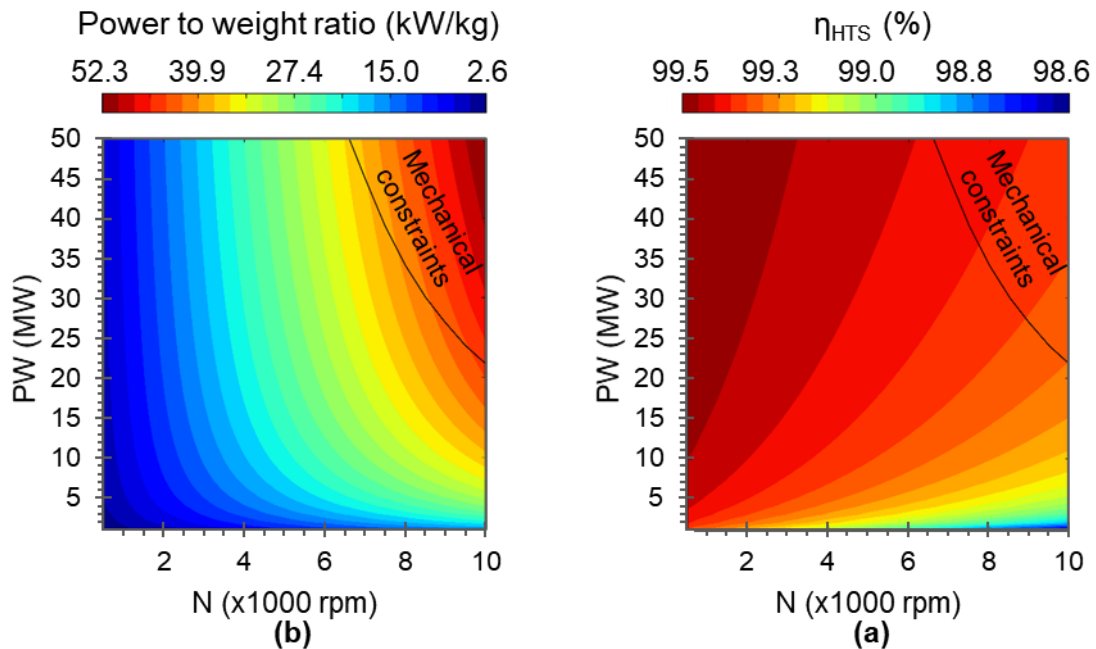


Figure 3-27 Power to weight ratio (a) and efficiency (b) against rated power and rotational speed

For the TeDP applications, the power to weight ratio of a single generator coupled with the LPT would typically vary between 25-35 kW/kg and the efficiency would be around 99.4%. For the motors, however, since all the power of the generator is distributed to several units, power to weight ratios are expected to be in the range of 15-25 kW/kg and an efficiency of 99.3%.

3.4.2.4 Design point and Off design point performance

The sizing and the efficiency estimation discussed that define the design point of the machine, apply for only a set of requirements of maximum rotational speed and power. In-flight operation, however, suggests that the actual operating conditions will be very different than the sizing ones, as power and rotational speed constantly vary. Therefore, predicting off-design performance of such a machine is of significant importance as it affects the operation of the gas turbine.

To achieve that, existing operating maps of motors can be used and scaled to reflect the size of the intended application. However, since HTS machines are still in experimental or limited application stage, such maps are not found in the literature. At the same time, any experimental data that are reported lack the extensive torque and rotational speed range that are needed for turbomachinery-based applications and occasionally include cooling losses as part of their reported efficiency.

Vratny et al. [126] estimated the off-design performance by using empirical correlations to scale individually the loss components listed in eq. (3-57), however, this approach can be applied for conventional technologies. The methodology that is used in this context is introduced by McDonald [135]. This is based a loss model proposed by Larminie and Lowry [136], where the losses are calculated based on the rotational speed, torque and a set of constants (C_0 - C_3) as shown in eq. (3-59).

$$PW_{loss} = C_0 + C_1\omega + C_2\omega^3 + C_3Q_R^2 \quad (3-59)$$

McDonald then by combining equations (3-44), (3-58) and (3-59) and the partial derivatives for torque and rotational speed of (3-58), expressed the coefficient

of the power loss model as shown in eq. (3-60) – (3-63), where k_0 is a user input constant between 0 and 1.

$$C_0 = k_0 \frac{\omega Q_R}{6} \frac{1-\eta}{\eta} \quad (3-60)$$

$$C_1 = \frac{-3C_0}{2\omega} + Q_R \frac{1-\eta}{4\eta} \quad (3-61)$$

$$C_2 = \frac{C_0}{2\omega^3} + Q_R \frac{1-\eta}{4\omega^2\eta} \quad (3-62)$$

$$C_3 = \omega \frac{1-\eta}{2Q_R\eta} \quad (3-63)$$

At design point, based on the power requirement, rotational speed and the resulting efficiency the four coefficients are calculated, and at off-design operation, equation (3-59) is used to estimate the efficiency.

3.4.2.5 Integration with gas turbine performance simulation software

In the context of a gas turbine simulation software, the electrical equipment, when included, needs to represent the transmission losses of the system. The performance of a motor or a generator is defined by the system of 4 equations ((3-44), (3-56), (3-58) and (3-59)) with 6 variables (PW , P_{Wloss} , η , f , Q_R and ω) and 6 constants (C_{0-3} , k_0 and p). Therefore, its performance can at all times be determined by the definition of any two of variables (apart from the combination of f and ω which are co-dependent). Apart from the system of equations discussed, connection ports are also required to transfer information between the connected components. Two ports are considered, one that transfers information about mechanical connections, such any two of power, torque or rotational speed (i.e. shafts), and another that transfers information about the electrical connections, such as power and current frequency. The electrical port sums the information regarding the power input or output of any connected components but maintains the frequency the same.

Both a generator and a motor component have mechanical and electrical ports however their difference comes on the power flow which is described by equations (3-64) and (3-65). Finally, the cable component is added to account for the associated losses, which are estimated based on a power loss to length ratio ($k_{loss,cable}$) and the cable length (l_{cable}), eq. (3-66), (3-67). For a high power HTS cable, 5 W/m are considered as in [67]. An example of the electrical transmission with the connected motors, generators and cable is presented in Figure 3-28.

$$PW_{Electrical} = PW \cdot \eta_{HTS} \quad \text{for generators} \quad (3-64)$$

$$PW_{Electrical} = \frac{PW}{\eta_{HTS}} \quad \text{for motors} \quad (3-65)$$

$$PW_{out} = PW_{in} - PW_{loss,cable} \quad (3-66)$$

$$PW_{loss,cable} = l_{cable} \cdot k_{loss,cable} \quad (3-67)$$

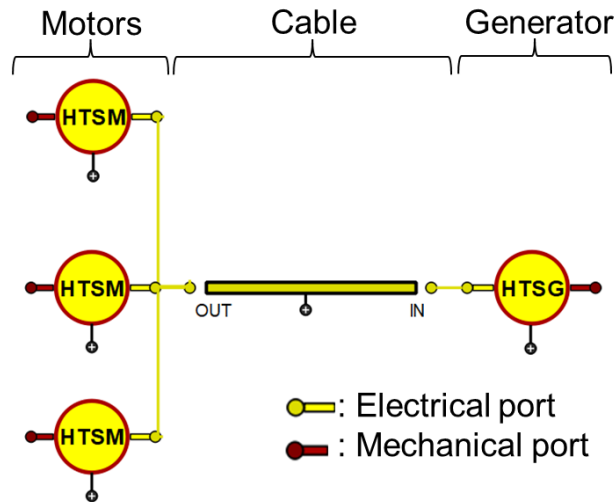


Figure 3-28 Electrical transmission assembly in PROOSIS

3.4.3 Propulsion system miscellaneous

3.4.3.1 Gearbox

If a gearbox needs to be included in the configuration, a constant efficiency is considered to account for the transmission losses and its weight is estimated based on the empirical correlations suggested by Hendricks et al. [137]. The weight is calculated based on a parametric value (PV_{GB}), eq. (3-68), which in turn is a function of the power input and the inlet and outlet rotational speeds, (3-69). This correlation has been derived by mapping existing designs from turboprop and turboshaft applications, although it does not include more recent designs it is considered be adequate for the purposes of the modelling. In the context of this study a gearbox, if needs to be used, should have performance characteristics that allow for a PV_{GB} greater than 1 to allow for a feasible design.

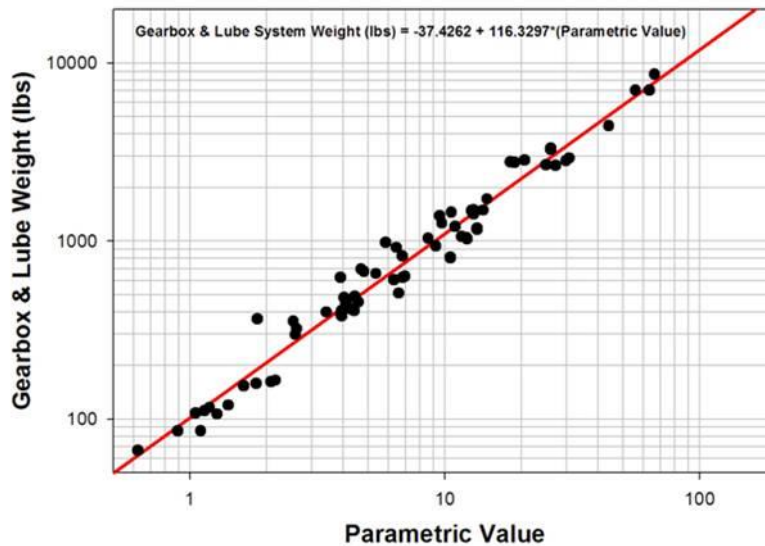


Figure 3-29 Gearbox including the lube weight as a function of its power and rotational speed, adopted from [137]

$$W_{Gearbox} = -37.4262 + 116.3297 \times PV_{GB} \quad (3-68)$$

$$PV_{GB} = \left(\frac{PW_{in}}{N_{out}} \right)^{0.75} \left(\frac{N_{in}}{N_{out}} \right)^{0.15} \quad (3-69)$$

3.4.3.2 Heat exchanger for fuel heating

To account for the fuel to air heat exchanger for fuel preheating the following model is considered. The top-level performance of a heat exchanger can be described by the heat balance of the two sides, cold and hot, and the definition of the effectiveness of the heat exchanger (EF_{HX}). The cold side represents the fuel-hydrogen stream and the hot side the exhaust of the LPT. The heat balance is described by eq. (3-70) based on the cold and hot side flow rate (W_{cold} , W_{hot}), the temperature drop of the two sides ($T_{hot,in} - T_{hot,out}$, $T_{cold,in} - T_{cold,out}$) and the mean specific heat capacities (Cp_{cold} , Cp_{hot}). The heat exchanger effectiveness is then calculated based on eq. (3-71) according to the actual heat transferred between the two sides (Q_{HX}), the maximum potential for heat transfer based on the temperature difference of the two inlets ($T_{hot,in} - T_{cold,in}$) and the minimum heat capacity rate C_{min} , determined by eq. (3-72). The Cp_{cold} and Cp_{hot} are calculated based on the mean Cp dictated by the corresponding inlet and outlet temperatures and the flow properties, i.e. pure hydrogen for the cold side and air and gas mixture on the hot side.

$$W_{hot} Cp_{hot} (T_{hot,in} - T_{hot,out}) = W_{cold} Cp_{cold} (T_{cold,out} - T_{cold,in}) = Q_{HX} \quad (3-70)$$

$$EF_{HX} = \frac{Q_{HX}}{C_{min} (T_{hot,in} - T_{cold,in})} \quad (3-71)$$

$$C_{min} = \min(Cp_{cold} W_{cold}, Cp_{hot} W_{hot}) \quad (3-72)$$

Design point sizing:

To size the heat exchanger (HX) as a standalone component, apart from the inlet conditions of the two sides (W_{cold} , W_{hot} , T_{cold} , T_{hot}) an additional input is required which can either be selected from EF_{HX} , $T_{cold,out}$ or $T_{hot,out}$. The former two are more convenient to be selected because they refer to the capabilities of a HX design selected or to the fuel injection temperature requirements.

Off design point calculations:

For the off design performance calculation of the HX, the EF_{HX} is used directly as an input. As a first order approximation its value is varied according eq. (3-73), discussed by Walsh and Fletcher [138], based on the ratio of cold side corrected mass flow ($W_{cold,corrected}$) to its DP value ($W_{cold,corrected@DP}$) and DP effectiveness ($EF_{HX,DP}$). This equation is proposed for intercoolers but for regenerators and recuperators the ratio of the actual mass flow to its DP value can be used [138]. As both the pressure and the temperature at the inlet of the cold side is approximately constant throughout the engine operating conditions the EF_{HX} is heavily influenced by the fuel flow demand.

$$EF_{HX,OD} = 1 - \frac{W_{cold,corrected} / W_{cold,corrected@DP}}{1 - EF_{HX,DP}} \quad (3-73)$$

To accommodate the HX performance in PROOSIS a new component was created, shown in Figure 3-30, that comprises the equations (3-70) to (3-73) and the fluid models were updated to account the properties of pure hydrogen flow.

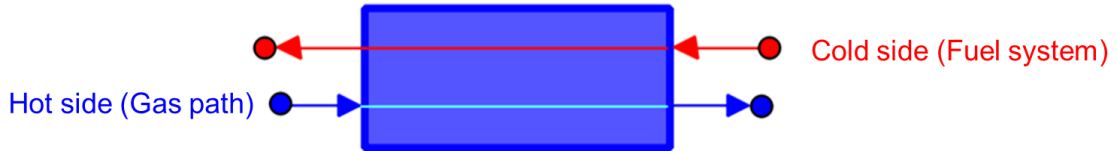


Figure 3-30 Heat exchanger component for fuel pre-heating created in PROOSIS

Weight consideration:

Detailed sizing of the heat exchanger is not addressed in the context of this work. However, in order to consider the added weight, the design from Brewer [20] is adopted as a baseline that featured approximately EF_{HX} 0.8, Q_{HX} 1MW at top of climb and weighed 77 kg. For the applications that will follow the weight of the HX (M_{HX}) is scaled linearly based on the reference HX and the Q_{HX} of the sized application, as shown in eq. (3-74).

$$M_{HX} = 77 \times 10^{-6} Q_{HX} \quad (3-74)$$

3.4.4 Turboelectric distributed propulsion model set up

To better discuss the model set-up of the TeDP configurations, first, the smaller subsystems and their design point and off-design are analysed. Then the integrated assembly is presented and discussed. The models are created within PROOSIS simulation environment based on existing components found in TURBO library [120] and the created ones discussed in sections 3.4.1, 3.4.2 and 3.4.3. The description of the models that will follow is, however, presented generalised so that it can be applied to any other equivalent gas turbine simulation software (e.g. TURBOMATCH, NPSS).

3.4.4.1 Propulsor model

The remote propulsor model comprises the BLI intake, the fan, the variable nozzle area and the electrical motor modules as shown in Figure 3-31.

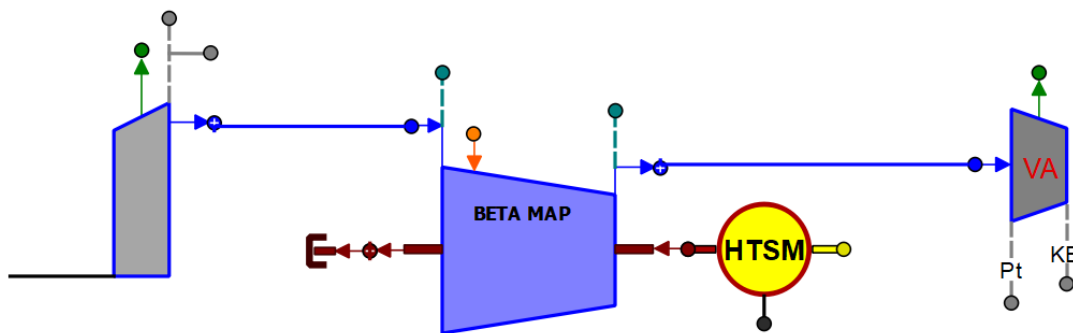


Figure 3-31 Remote fan module with BLI intake, VAN and motor components

Design Point mathematical model

For the design point calculations, the following parameters that fully describe the sized application need to be sized:

Parameters to be sized:

1. BLI inlet control width (W_{BL})
2. Fan map scaling factors that involve the:
 - a. Efficiency (η_{is})
 - b. Corrected mass flow (W_{cor})

- c. Pressure ratio (PR)
- d. Rotational speed (N)
- 3. Nozzle exit area (A_{BLI})
- 4. Corrected mass flow area at nozzle inlet ($W_{cor,noz}$)

Variable used as inputs:

- 1. Net propulsive force (NPF)
- 2. Fan performance characteristics:
 - a. Efficiency (η_{is})
 - b. Pressure ratio (FPR_{BLI})
 - c. Rotational speed (N_{BLI})
 - d. Operating point position on baseline map
- 3. Corrected mass flow ratio relative to its design value at nozzle inlet ($W_{cor, ratio}$)

This selection of input variables forms a system of non-linear equations, the solution of which requires the selection of one variable that needs to be initialised and iterated until the solution converges. For that reason, the inlet mass flow is selected.

Off Design mathematical model

For an off-design simulation, the parameters calculated at design point are used. Only a single variable is required to be used as a control input and in the context of this work the following is considered:

- NPF or input power

Rotational speed is not required in this case since it is dictated by the area of the VAN. Similarly as before some variables are also selected to be initialized and iterated. These are:

- Mass flow (W_{BLI})
- Fan rotational speed (N_{BLI})

3.4.4.2 Turboshaft model

The turboshaft model considered has a two spool configuration with a free turbine that powers the electrical generator. Several bleeds are utilised from the exit of the intermediate pressure compressor (IPC) to the exit of the high pressure compressor (HPC) for turbine cooling use. The heat exchanger at the exhaust of the power turbine extracts heat from the main gas path and increases the fuel temperature before injection to the combustion chamber (CC). The schematic of the model is presented in Figure 3-32.

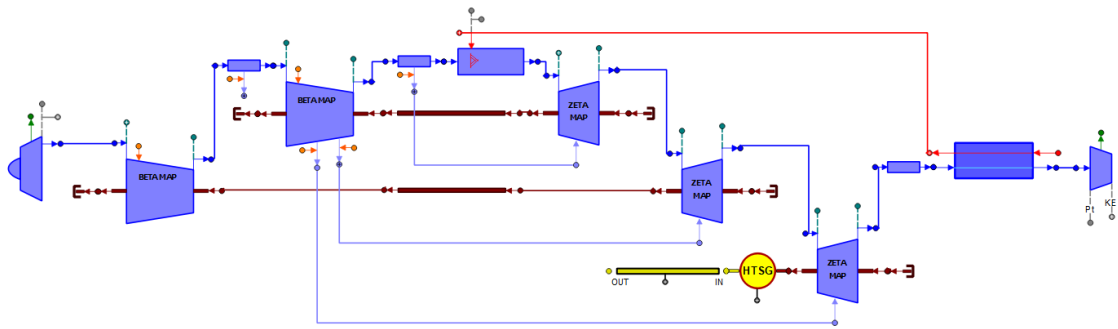


Figure 3-32 Two spool turboshaft with free power turbine and fuel to air heat exchanger

Design Point mathematical model

For the turboshaft design point calculations the following parameters that fully describe the sized application need to be sized:

1. Compressor and turbine map scaling factors that involve the:
 - a. Efficiency (η_{is})
 - b. Corrected mass flow (W_{cor})
 - c. Pressure ratio (PR)
 - d. Rotational speed (N)
2. Core nozzle exit area (A_{core})

Variables used as inputs:

1. Net thrust of the core (FN_{core})
2. LPT power output (PW_{LPT})

3. Compressors' performance characteristics:
 - a. Efficiency (η_{is})
 - b. Pressure ratio (PR)
 - c. Operating point position on baseline map
4. Turbines performance characteristics:
 - a. Efficiency (η_{is})
 - b. Operating point position on baseline map
5. Turbine entry temperature (TET)
6. Shafts rotational speed

For this selection of input variables, the variables that are selected to be initialized and iterated are:

- Inlet mass flow rate (W)
- Fuel to air ratio at the exit of the CC

Off Design mathematical model

Using the parameters calculated at the design point, the off design calculation requires three control variables.

1. Net thrust (FN)
2. Generator power output (PW_G)
3. Free turbine rotational speed (N_{LPT})

In case fuel temperature and pressure at the inlet of the heat exchanger are not set at a constant value, then they both need to be defined.

Iteration variables are:

- Inlet mass flow rate (W)
- Fuel to air ratio at the exit of the CC
- Shafts rotational speeds (N_{HP} , N_{IP})

3.4.4.3 Turbofan model

The turbofan model has exactly the same configuration as the turboshaft model with the only addition the fan. The fan allows for different performance between

the core and by-pass streams and is connected to the LPT via a gearbox. The latter is included to simulate the reduction of the rotational speed as well as transmission losses. The schematic of the Turbofan is presented in Figure 3-33.

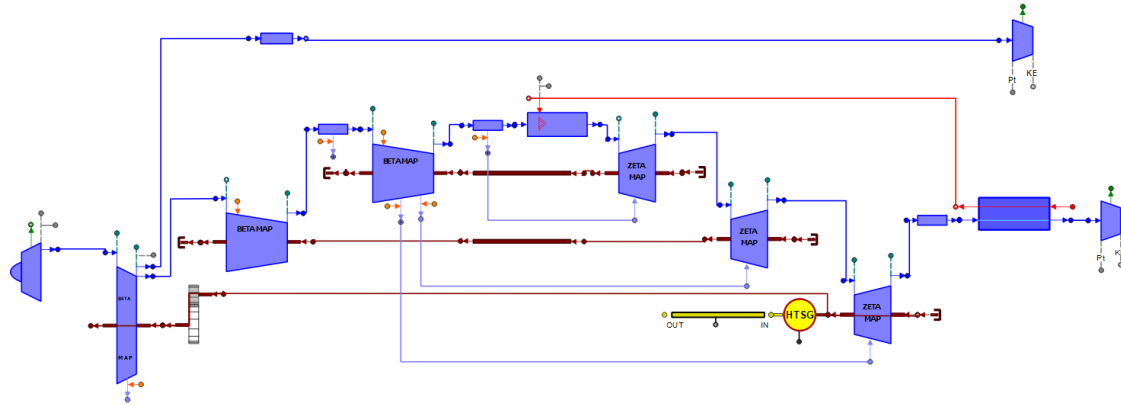


Figure 3-33 Three spool turbofan with generator connected to LPT and fuel to air heat exchanger

Design Point mathematical model

The turbofan parameters that need to be sized are the same ones as the turboshaft design point and some additional ones related to the added fan:

1. Fan map scaling factors for the core and by-pass streams that involve the:
 - a. Efficiency (η_{is})
 - b. Corrected mass flow (W_{cor})
 - c. Pressure ratio (FPR)
 - d. Rotational speed (N)
2. Bypass nozzle exit area (A_{BP})
3. Gearbox transmission ratio

Variables used as inputs are the same as the turboshaft case and additional ones are considered:

1. Net thrust of the bypass system (FN_{BP})
2. Fan performance characteristics:
 - a. Efficiency (η_{is})

- b. Pressure ratio (FPR_{TF})
- c. Operating point position on baseline map

Compared to the turboshaft case, the by-pass ratio (BPR) needs to also be included in the iteration variables.

Off Design mathematical model

The off-design calculation requires the same three control variables as the turboshaft model, where net thrust refers to the net thrust of the engine.

Additionally to the iteration variables discussed in the turboshaft model the BPR needs to be included as well.

3.4.4.4 Turbo-electric distributed propulsion model set up

The distributed propulsion model is built by combining the turboshaft model with several propulsors. Since every fan operates under different conditions, an equivalent number of BLI fan modules are used depending on the number of fans of the intended application. An example of a TeDP system modelled in PROOSIS is shown in Figure 3-34.

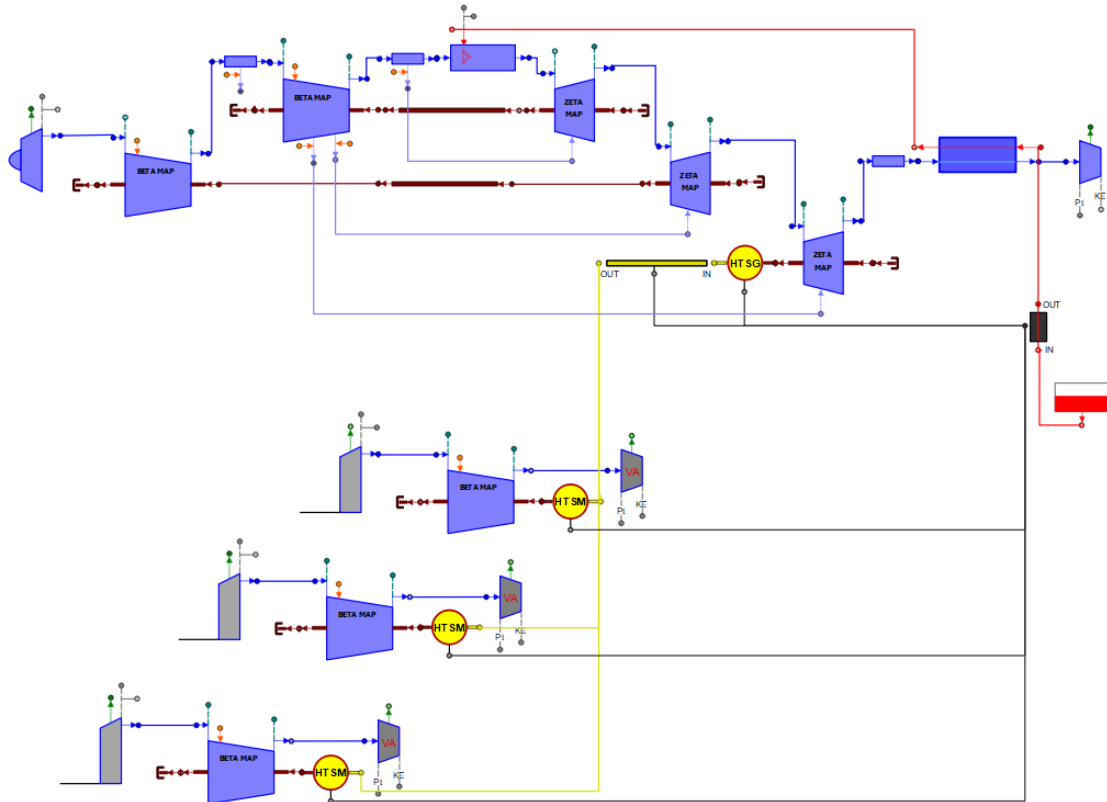


Figure 3-34 Two spool turboshaft with free power turbine and fuel to air heat exchanger

The turboshaft produces a small amount of thrust and produces power in the free turbine. Through the electrical transmission, the power generated is distributed to the fans according to their power requirement. The current frequency in the transmission system is the same, which leads to a synchronous transmission between the power turbine and the remote fans. However, the rotational speed of those two does not have to match. In this study, it is introduced that the two rotational speeds although synchronous, can differ according to the pole pairs of the generator and the motors. In such a way

a transmission ratio is considered and determined by the pole pair ratio of the generator and the motors, as is described by eq. (3-75).

$$f = \frac{P_{generator} \omega_{generator}}{2\pi} = \frac{P_{motor} \omega_{motor}}{2\pi} \quad \text{or} \quad \frac{N_{motor}}{N_{generator}} = \frac{P_{generator}}{P_{motor}} \quad (3-75)$$

To account for the fuel heating and the cooling of the superconducting systems, it is assumed that all the heat rejected by the electrical machines is transferred to the cold LH₂ that is coming from the tanks. This is a first-order approximation to determine whether the heat sources available can adequately pre-heat the LH₂. Afterwards, the warmer hydrogen is introduced to the fuel to air heat exchanger for further heating before injection in the CC.

3.4.4.5 Integrated model Design Point calculation

The design point calculation of the integrated configuration, apart from determining the parameters discussed in the previous subsections, aims also at defining the schedule of the remote propulsors' nozzle area.

Parameters to be sized:

Fan module related applicable to the whole array

1. BLI inlet control width (W_{BL})
2. Fan map scaling factors that involve the:
 - a. Efficiency
 - b. Corrected mass flow
 - c. Pressure ratio
 - d. Rotational speed
3. Nozzle exit area (A_{BLI})
4. Corrected mass flow area at nozzle inlet (W_{cor,BLI_DP})

Turboshaft related:

5. Compressor and turbine map scaling factors that involve the:
 - a. Efficiency (η_{is})
 - b. Corrected mass flow (W_{corTS_DP})
 - c. Pressure ratio (PR)

- d. Rotational speed (N_{HP} , N_{IP})
- 6. Core nozzle exit area (A_{core})

Variable used as inputs:

Fan module related

- 1. Net thrust of the whole array ($F_{N_{array}}$)
- 2. Fan closest to the centreline performance characteristics:
 - a. Efficiency (η_{is})
 - b. Pressure ratio (FPR_{BLI})
 - c. Rotational speed (N_{BLI})
 - d. Operating point position on baseline map
- 3. Corrected mass flow ratio relative to its design value at nozzle inlet

Turboshaft related:

- 4. Net thrust of the core ($F_{N_{core}}$)
- 5. Compressors performance characteristics:
 - a. Efficiency (η_{is})
 - b. Pressure ratio (PR)
 - c. Operating point position on baseline map
- 6. Turbines performance characteristics:
 - a. Efficiency (η_{is})
 - b. Operating point position on baseline map
- 7. Turbine entry temperature (TET)
- 8. Shafts rotational speed (N_{HP} , N_{IP})

Regarding the remote fan array, the input variables listed only determine the fan closest to the centreline of the aircraft. Previous studies suggest that [55, 59] all the fans are sized based on the same inputs (i.e. pressure ratio and efficiency). At the same time it could be argued that the thrust split between the fans can be optimally distributed to maximise the benefits of the BLI as demonstrated in [27]. However, both approaches lead to slightly different sizing on the fans, since the inlet flow conditions differ. In the context of this work and it is

considered that all the are sized identically. For the design point calculation, this translates that all the fans share the same:

1. BL width (w_{BL})
2. Nacelle dimensions
3. Fan size and performance (exactly the same operating maps)
4. Exit nozzle area and schedule

Effectively at design point, only the fan closest to the centreline will be operating at design point conditions. The rest, due to the differences in the positioning over the fuselage, would operate at off-design with slightly different performance and NPF.

To incorporate this assumption to the design point calculation the following work flow is used. At first the fan closest to the centre line is sized based on an initial value of NPF. Next the sized parameters sized are set to the rest of the propulsors and their individual NPF is estimated. The resulting total net thrust produced is compared to the target input value and this process is iterated until convergence is reached. Schematically the routine is shown in Figure 3-35.

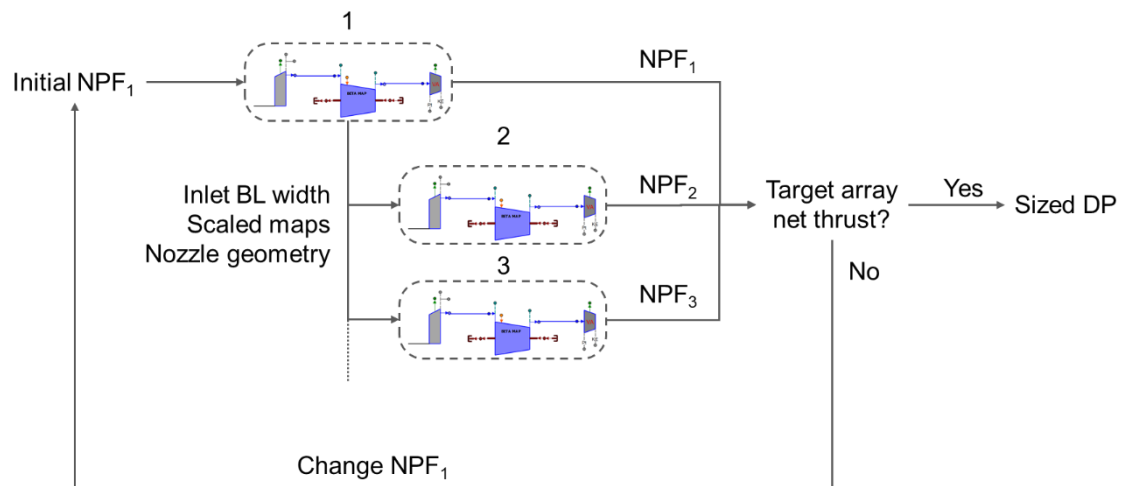


Figure 3-35 BLI Fan array sizing loop

VAN schedule:

After the design point calculation is completed, the VAN schedule is automatically created using the process discussed in 3.4.1.2.

3.4.4.6 Integrated model off-design point simulation

For the off-design simulation, the following are set up for the calculation:

- pressure losses in the ducts, combustion chamber and in both streams of the heat exchanger are set to vary with the corrected mass flow rate at the corresponding inlet [138].
- burner efficiency is a function of the burner loading [120].
- heat exchanger effectiveness is varied according to the corrected mass flow of the cold side, as discussed in section 3.4.1.2.

Variables used as inputs:

1. The overall net thrust of the propulsion system (FN_{tot})

Iteration variables:

1. Core mass flow (W_{core})
2. Mass flow of every BLI fan (W_{BLI})
3. Fuel to air ratio at the CC exit
4. All three-shaft rotational speeds (N_{IP} , N_{HP} , N_{LP})

3.4.5 Turbomachinery weight estimation

The turbomachinery weight estimation is a significant aspect of the propulsion system sizing. Several weight estimation methods have been developed over the years and a thorough review of those has been conducted by Lolis et al. [139]. The methods can be divided into “whole engine” approach and component-based.

“Whole engine” based approaches effectively estimate the propulsion system weight on a single empirical correlation according to key design characteristics. Whitehead et al. [140] suggested that the engine weight is a function of the inlet mass flow at its design point, however, this approach neglects state of the art design features such as OPR, FPR and BPR. Svoboda [141] correlated the engine weight according to its rated thrust, but similarly as before the effect of the fan size and BPR are not accounted for. Jenkinson [101] and Raymer [102] apart from the rated thrust included the BPR in their derived correlations and Torenbeek [106] added the inlet mass flow. All of these methods, however, refer to conventional turbofan architectures and have been based on engine data that are mostly outdated and are not suitable for distributed propulsion applications. In addition, the main engines produce both power and thrust and that combined feature cannot be captured by the methods discussed.

The component-based approach is then considered to be more appropriate as the propulsion system weight can be estimated by summing the individual subsystems and turbomachinery components. Notably, Sagerser et al at first developed such a methodology and the component weights were estimated based on physical rules combined with available engine data correlations. Another method, introduced by NASA, named WATE, increased the complexity of the weight estimation by breaking down the components into subcomponents and used analytical models to estimate the weights. The weight of a compressor for example is considered to be the sum of the rotor and stator blades, mounting discs and casing. These methods require a substantially higher amount of inputs related to performance, geometry and flow characteristics as well as a preliminary sizing of the gas path of the engine. At

the same time, they offer higher accuracy than the “whole engine” approach [139] in the order of $\pm 10\%$ but this error can even further decrease to $\pm 5\%$ if assumed inputs are minimised.

Lolis recognising the potential of such approaches and due to the difficulty of acquiring these tools, developed within Cranfield University a component-based weight estimation tool called ATLAS based on the same principles. In the context of this study, this will be used as it can accurately quantify the changes in component weight the design space exploration imposes. A complete description of the methodology and the functionality of the tool is thoroughly addressed in [142]. Herein, however, a shorter analysis is provided and inputs and sizing strategies are discussed.

The weight estimation tool has two distinct functions, the gas path sizing and the weight estimation of the component assembly. The latter is slightly more computationally demanding and requires first the completion of the former. The user inputs can be divided into three distinctive categories for each component, the material and flow properties, performance characteristics and key geometry features.

Material and flow properties:

Material selection and their properties are not considered in depth in this study, instead typical and recommended ones are used. The gas path flow properties are based on burnt kerosene and air mixture and do not reflect the corresponding hydrogen ones. Although this is a drawback of the tool, it only affects the post-combustion turbomachinery components. It is therefore considered that the final geometry and therefore weight is not going to substantially differ for the preliminary stages of the propulsion system design.

Performance characteristics

For the sizing of the gas path, ATLAS requires a set of inputs for every turbomachinery component both at the inlet and outlet. For both compressors and turbines, these refer to:

1. Total temperature and pressure
2. Mass flow
3. Flow humidity
4. Characteristic speedline

The speedline is included in case the compressors and turbines are sized based on different operating points. Additionally, the fan component also requires the bypass ratio and the turbines the fuel to air ratio.

Key geometry features:

Apart from the performance characteristics, every component requires some additional flow properties and geometry characteristics related both at the inlet and outlet as well as key features of every component. Although detailed information can be included in the sizing (compressor, turbines and combustor configuration, rotor and stator spacing to chord ratio, rotor to stator spacing, maximum stage loadings etc.), herein only the parameters that affect the design are discussed.

Fan both for bypass and core gas path:

1. Axial Mach number (inlet and outlet)
2. Flow angle (inlet and outlet)
3. Hub to tip ratio (inlet)
4. Mean line evolution

Compressors and turbines of the gas generator:

1. Axial Mach number (inlet and outlet)
2. Flow angle (inlet and outlet)
3. Hub to tip ratio (inlet)
4. Mean line evolution
5. Number of stages

Low pressure turbine:

1. Axial Mach number (inlet and outlet)
2. Flow angle (inlet and outlet)

3. Hub to tip ratio (outlet)
4. Mean line evolution

Mach numbers and hub to tip ratios are preferably used since they are independent of the performance characteristics discussed and can readily be assumed from similar applications. For this preliminary sizing, the number of stages of the gas generator components is used as an input since a combination of stage loadings with performance requirements can lead to conservative results. In an in-depth approach, several iterations between the performance calculations and the gas path sizing are required in order to match pressure ratio and efficiency characteristics with the number of stages, stage loadings and rotational speeds. This, however, is not addressed in this work. For the LPT the number of stages is dictated by the tool to capture the weight change associated with the power generation requirements.

Assembly and rotational speed selection:

In addition to the geometry inputs, the layout of the engine needs to be provided so that the connections between components are determined and information about common rotational speeds is shared. The rotational speeds are then selected based on aerodynamic and mechanical limits imposed. Rotational speed for compressors and fans is restricted by a maximum velocity and a relative Mach number at the highest tip across the geometry. For turbines in addition to the limits discussed for compressors, the area loading ($A \cdot N^2$) is also considered. For a compressor coupled with a turbine the rotational speed is automatically selected as the lowest of the ones imposed by the combined limits.

3.4.6 Turboelectric distributed propulsion system multidisciplinary sizing framework

The multidisciplinary sizing framework aims to combine all the propulsion system-related technologies discussed, create a performance model and estimate its overall weight. In addition, two aspects are tackled as well, the identification of multi-point sizing requirements of the different technologies and the narrowing down of the available design variables.

3.4.6.1 Multi-point sizing requirements of coupled components

For the first aspect, the sizing requirements of the components are determined by the maximum operating conditions throughout the flight envelope. Two are identified, the top of climb (ToC), where the turbomachinery components operate at their maximum non-dimensional power setting, and the end of runway (EoR), where maximum measures can be observed such as maximum power requirements and temperatures along the gas path. For all the components a summary of those is presented in Table 3-9.

Table 3-9 Propulsion system components maximum requirements occurrence

	Max power setting	Max rotational speed	Max Power output / requirement	Max exhaust temperature
Remote fans module	ToC	ToC / EoR	EoR	-
Compressors	ToC	ToC / EoR	EoR	EoR
Combustion chamber	-	-	-	EoR
Turbines	ToC	ToC / EoR	EoR	-
Motors / Generators	-	ToC	EoR	-

First of all the maximum power setting of all the turbomachinery components is usually reached at top of climb and therefore these conditions are set for the design point sizing. Regarding the weight, a combination of maximum power setting and maximum rotational speed needs to be considered. The latter also

needs to be carefully selected to comply with the mechanical and aerodynamic limits. Usually, the maximum rotational speed is met at EoR but in the context of this work, ToC is observed to have a higher rotational speed. Therefore, the gas path geometry and weight estimation of the fans, compressors and turbines are based on the performance characteristics met at ToC.

Additionally, even though at ToC the OPR is higher, the maximum TET, as well as compressor exhaust temperature, are met at EoR due to the higher ambient temperature and pressure. As such the selection of TET and OPR design point need to take into account these limits.

The motors and generators are designed based on a rated power and rotational speed. Since they are coupled with the turbomachinery components where corresponding maximum power and rotational speed are met during different operating conditions, they are sized based on a combination of these two to ensure adequate performance throughout.

3.4.6.2 Propulsion system design degrees of freedom

In section 3.4.4.5, the input variables required for the design point calculation are discussed. Some of them can be constrained or semi-constrained, others can be co-dependent and others can be freely chosen. To minimise the complexity, the framework aims to determine the variables from the former two categories and minimise the essential “free” ones which would be considered to as the degrees of freedom of the design space. All of the variables of the design point are listed and categorised in Table 3-10.

Number of propulsors: Degree of freedom, since the same thrust can be achieved by utilising different numbers of fans. However, as a secondary consideration, integration aspects need to be taken into account based on the available installation space.

Fan array net thrust: Although the net thrust of the propulsion system is dictated by the requirements of the aircraft, the net thrust of the array is co-dependent with the net thrust produced by the core.

Fan pressure ratio: Degree of freedom. Same thrust levels can be achieved with different fan pressure ratios

Table 3-10 Design point input variables

Fan Array related	Number of fans	Degree of freedom
	Fan array net thrust	Co-dependent
	Pressure ratio	Degree of freedom
	Efficiency	Constrained
Turboshaft related	Core net thrust	Co- dependent
	Overall Pressure Ratio	Constrained & Co- dependent
	Turbomachinery Efficiencies	Constrained
	Turbine Inlet temperature	Constrained & Co- dependent
	Shaft rotational speeds	Constrained & Co- dependent

Fan efficiency: Fan efficiency is constrained by aerodynamic limits and technology levels and is hence a direct input.

Core net thrust: Co-dependent on the net thrust produced by the core. The thrust split effectively

Overall pressure ratio: Constrained by aerodynamic limits and temperature at the exit of the HPC.

Compressor and turbine efficiencies: Constrained by aerodynamic limits and technology levels and is hence direct inputs.

TET: Constrained by the operating temperature limits of the materials used at the first stage of the HPT.

Shaft rotational speeds: Constrained by centrifugal forces acting on the rotating parts and potentially co-dependent between connected shafts. They do not affect the DP and OD calculations (as these are non-dimensionalised and normalised according to a reference value) but they have a significant impact on the weight estimation of both the turbomachinery and electrical components.

3.4.6.3 Multidisciplinary sizing of turboshaft-powered TeDP configurations

The framework sizes all the constrained and co-depended variables based on a set of inputs of the degrees of freedom. It follows a two-point design approach based on equivalent conditions for ToC, set as DP, and EoR set as OD. A third point to represent cruise conditions is not included as every design produced by the framework is later assessed based on overall mission level performance, which inherently includes the whole cruise segment. The framework comprises the engine performance simulation software, PROOSIS, the turbomachinery gas path sizing and weight estimation, ATLAS, and the electrical component weight estimation discussed in section 3.4.2.1. Schematically the process is presented in Figure 3-36.

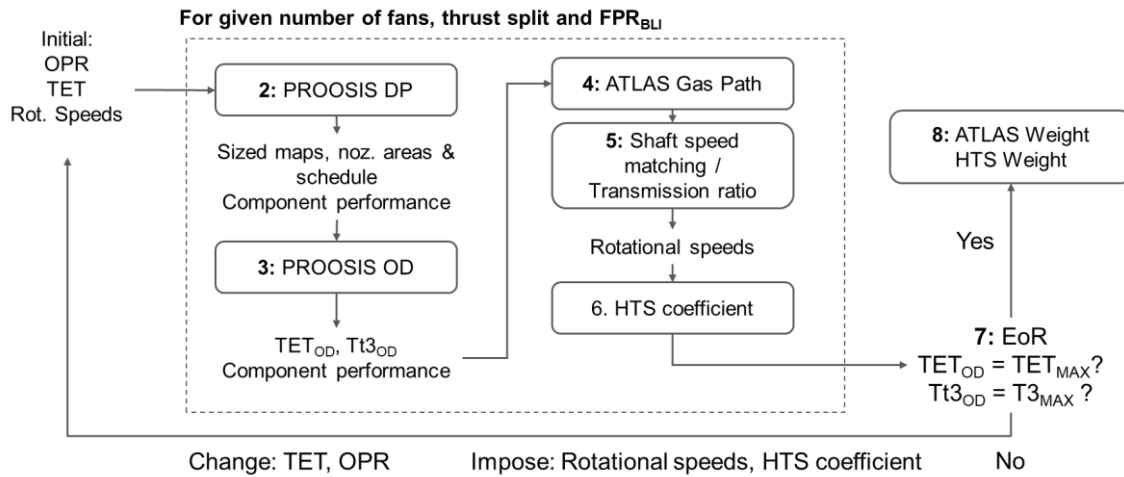


Figure 3-36 Propulsion system sizing and weight estimation loop

For a specified **number of fans**, **thrust split** between the fan array and the core as well as **FPR_{BLI}** the process is broken down in the following steps:

Step 1: An initial value of OPR, TET and rotational speeds is assumed.

Step 2: Based on the set values the DP is simulated and the component maps and performance, as well as nozzle areas and schedule, are calculated.

Step 3: The model runs at OD at the end of runway conditions and the component performance, maximum TET_{OD}, and HPC exhaust temperature (T3_{OD}) are calculated.

Step 4: Based on the performance of both DP and OD, ATLAS sizes the gas path and calculates the rotational speeds individually for the turboshaft components and the BLI fans as if they were not coupled.

Step 5: According to the individual rotational speeds the user selects whether the maximum limits of the LPT or the BLI fan will be matched. Ideally, the maximum rotational speeds of both the LPT and the remote fans could be matched if the transmission ratio can be freely chosen. In this case, however, it is a function of generator and motor pole pairs and can therefore only take discrete values such as $\frac{1}{2}$, $\frac{2}{3}$, $\frac{3}{4}$ etc. If the rotational speed limits are imposed by the LPT the inequality of (3-76) is used and the pole ratio closest to the maximum rotational speed ratio is chosen. Alternatively, if the speed limit is imposed by the BLI fan, inequality of (3-77) is used and the pole ratio is selected similarly.

$$\text{LPT sizing: } \frac{N_{Fan,max}}{N_{LPT,max}} \geq \frac{p_{generator}}{p_{motor}} \quad (3-76)$$

$$\text{Fan sizing: } \frac{N_{Fan,max}}{N_{LPT,max}} \leq \frac{p_{generator}}{p_{motor}} \quad (3-77)$$

Step 6: According to all the rotational speeds the motor and generator factors are estimated based on the power requirement calculated.

Step 7: If the EoR TET and T3 do not match the limits the process is iterated by correcting the design point TET, OPR and imposing the rotational speeds calculated as well as the HTS factors.

Step 8: Once the process has converged and the optimum TET and OPR have been selected, ATLAS calculates the weight of the turbomachinery components and the HTS sizing module the weight of the motors and generators. Since all the motors are identically sized their rated power is determined by the fan with the highest power requirement.

It is noted that although the

3.4.6.4 Adapted sizing approach for turbofan-powered TeDP configurations

Including a turbofan in the configuration increases the complexity since extra degrees of freedom for the design space exploration are added. These are related to the fan sizing of the fan of the main engine. In the turboshaft case, two degrees of freedom are recognised the number of fans as well as the fan pressure ratio. In this case, the fan pressure ratio of the fan of the main engine as well as the thrust split between the fan array and the by-pass system needs also to be considered and increases the degrees of freedom to four.

1. Number of remote fans
2. Remote fan pressure ratio (FPR_{BLI})
3. Engine fan pressure ratio (FPR_{TF})
4. Thrust split between engine fan and remote fan array

For the sizing and weight estimation, the framework is adapted for the turbofan model and two major modifications are introduced. First, the framework uses the four degrees of freedom described as inputs. Secondly, **step 5** is changed to accommodate the rotational speed of the fan of the main engine. Adding the fan to the LP system adds another limiting parameter to the rotational speed selected which is its aerodynamic speed limit. The fan is connected via a gearbox to the LPT and therefore its reduction ratio can be freely selected to match that maximum speed. However, if the combination of the by-pass net thrust and the engine fan pressure ratio reaches a limit, a gearbox is not used and the engine fan is driven directly by the LPT. To accommodate this constraint the following workflow is added to the framework.

At first, Atlas estimates the gas path geometry of the engine fan, the LPT and the BLI fans individually as if they are not coupled. Their maximum rotational speed is dictated by the corresponding aerodynamic limits. As in the turboshaft case, the electrical transmission ratio is established based on eq. (3-76) if the LPT rotational speed limit is matched or eq. (3-77) if the corresponding limit for the BLI fan is used. Afterwards, it is checked whether a gearbox needs to be

used. Based on the performance of the engine fan, its rotational speeds and the rotational speed of the LPT, the gear ratio and the gearbox parametric value (PV_{GB}) of eq. (3-69) are calculated. If the gear ratio and the PV_{GB} are bigger than 1.1 and 1.0 respectively, then the gearbox can be used as it lies in the feasible design space shown in Figure 3-29. Alternatively, the engine fan is directly driven by the LPT and the common rotational speed is selected as the minimum between the N_{LP} and the $N_{TF,max}$.

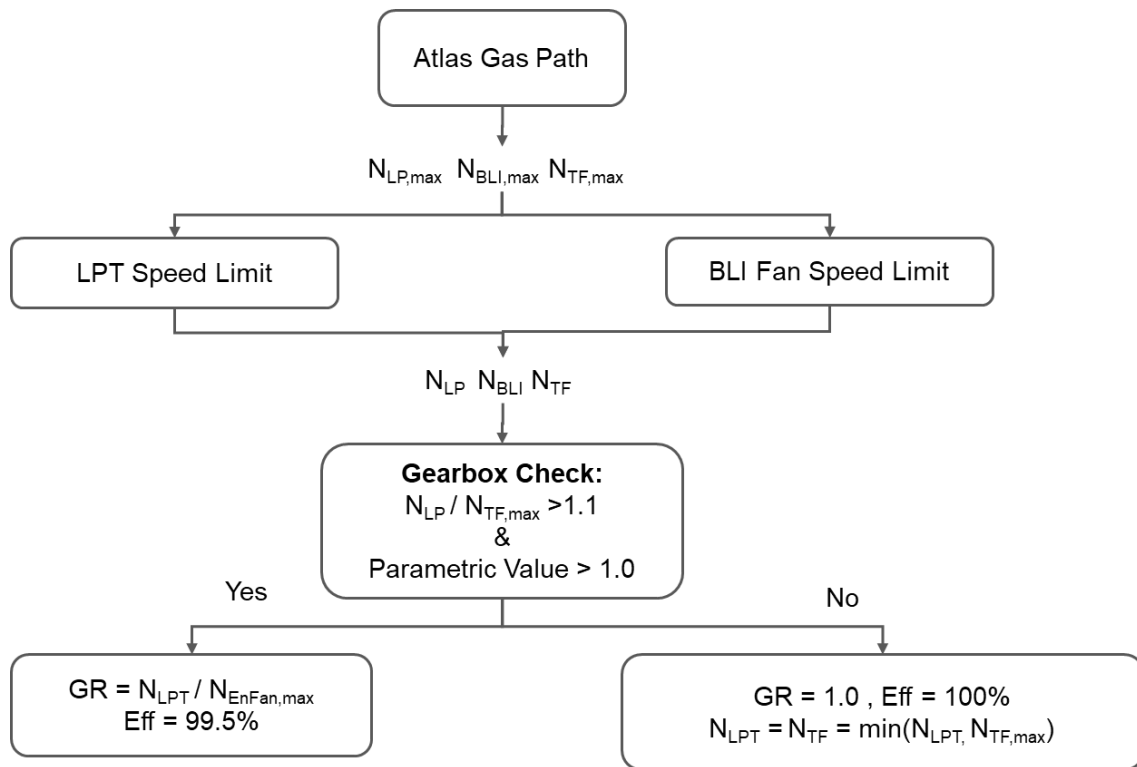


Figure 3-37 Rotational speed selection on turbofan based distributed propulsion

4 Results & Analysis

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This chapter includes the sizing results of the LH₂ powered BWB. At first, the design mission requirements are presented and then the aircraft geometry is discussed. Based on the available fuselage volume, the LH₂ tanks are then sized for two different cases, with and without pressure control through GH₂ extraction and a final layout is proposed. According to the LH₂ available on board, a design space exploration is performed for the TeDP and partially TeDP configurations for different numbers of fans. For every propulsion system reported a different aircraft model is sized and its range is assessed. Finally, the configuration that allows for the greater range is selected and the corresponding aircraft's characteristics and performance are thoroughly presented.

4.1 Aircraft requirements

The application selected aims to project a hydrogen-powered long-range application into 2050. The aircraft is sized based on some fundamental requirements that are considered to be representative of a state of the art aircraft that is predicted to enter service in 2024, the Boeing 777-9X [143]. As adopted by ENABLEH2 [11, 13, 31] the aircraft requirements are summarised in Table 4-1.

For the design mission, the aircraft needs to achieve at least a range of 7500 nmi while carrying 414 passengers in a two-class configuration. The payload is 46.7t and 68.7t for the design and the maximum payload missions respectively. From an operational point of view, the cruising altitude needs to be at least 33000 ft with a speed higher than Mach 0.82. An important parameter that drives the engine thrust requirements is that the total climb segment should be approximately 25 mins. Last but not least, the aircraft needs to comply with Code E airport compatibility limits, which means that the wingspan cannot exceed 65 m. The 777-9X for example has a wingspan of 71.8m but utilises folding wing tips, which when folded reduce the wingspan to 65m.

Table 4-1 aircraft sizing requirements

Requirements	Units	Values
Range (at least)	nmi	7500
PAX (Business & Economy)	-	414 (60 & 354)
Operational Stage length	nmi	3000
Typical Passenger Weight /	kg / t	115 / 47.61
Passenger Weight for Max Payload /	kg / t	166 / 68.71
Maximum time to climb (from 1500 ft, ISA)	min	25
Minimum Cruise Altitude	ft	33000
Minimum Design Cruise Mach Number	-	0.82
Maximum Cruise Altitude	ft	41000
Airports Compatibility Limits / Maximum	- / m	Code E / 65

4.2 Aircraft geometry

4.2.1 Baseline architecture

The baseline airframe considered is the NASA BWB kerosene powered concept proposed by Felder [28, 55, 122] and Kim [144, 145] namely the N3-X. This architecture underwent several iterations [40] until its final geometry was established, and two variants were presented. The first, designated N3-X featured a turbo-electric distributed propulsion configuration, where a set of 14 fans were powered by two wingtip-mounted turboshaft engines via a superconducting electrical transmission system. The second, namely N3-A, featured exactly the same geometry but with winglets serving also as vertical stabilisers and the thrust was provided by two turbofan engines mounted above the aft fuselage section. The two models are depicted in Figure 4-1.

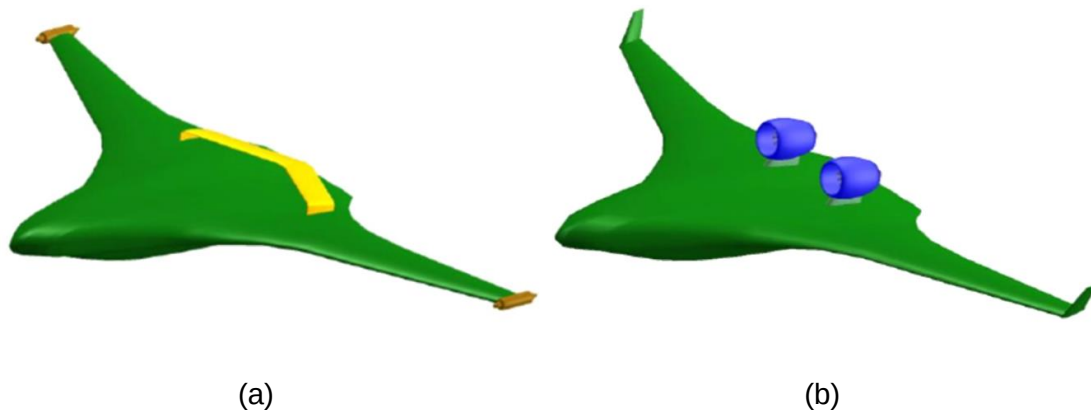


Figure 4-1 N3-X (a) and N3-A (b) courtesy of [55]

By assuming technology levels of 2035, the aircraft performance as discussed in [55], is summarised in Table 4-2. The fuel required for the maximum payload mission accounts for 18% of MTOW. Comparing the fuel consumption with an equivalent aircraft that was operational at the same time these results were produced, Felder claimed 72% fuel savings. This difference is attributed to the more efficient technology levels adopted, both in terms of engine cycle as well as the distributed propulsion featuring BLI fans, but most importantly to the more efficient airframe. Another feature of the aircraft that stands out is that its

fuselage is relatively short, 42m, due to the multi-bay layout of the cabin but maintains the wingspan at 65m.

Table 4-2 N3A and N3-X specifications [28]

Requirements	Units	Values	
Aircraft	-	N3A	N3-X
Range	nmi	7500	
Number of Passengers	-	300	
OWE	t	129.3	121.3
Payload	t	53.6	
On-board fuel (Jet A-1)	t	67	39.9
MTOW	t	249.9	214.8
Fuselage length	m	42	
Wing span	m	65	

4.2.2 Geometry sizing

As discussed, the N3-X that utilises Jet A-1 can accommodate 300 passengers in a two-class configuration. The application that is considered here has an increased passenger capacity of 414 and needs to use LH₂ as fuel. As such the airframe sizing needs to take both factors into account. As demonstrated by Goldberg [27], simply switching fuels and attempting to retrofit a Jet A-1 fuelled aircraft to utilise LH₂ would not suffice, due to the restricted volume available for LH₂ storage.

For the sizing, it is considered that the outer wing box of the N3-X is going to be kept unchanged to restrict the wingspan of the scaled aircraft to 65m. Only the fuselage section would be linearly scaled in length and depth, while maintaining the thickness to chord distribution, in a similar way as the T&W family aircraft are developed. Although further optimisation of the new geometry of the aircraft can be performed and more complex aspects such as centre of gravity shift throughout the flight need to be examined, these are not addressed at this preliminary stage and are considered to be outside of the scope of this work.

4.2.2.1 Cabin sizing

At first, the passenger cabin area is determined both for the baseline as well as the scaled aircraft. Both are calculated based on the number of passengers, their division among the two classes, the number of bays as well as the sweep angle. In both cases, the sweep angle is the same and for the scaled geometry 4 bays are selected as well since more would lead to a widened fuselage. In summary, inputs and results for both cabins are presented in Table 4-3. As both cabins share the abreast seating arrangement per bay, as well as the same number of bays, their individual and the total width is the same. The scaled cabin as it carries more passengers requires more closets, galleys and lavatories and overall requires a 24.4% bigger planform area which subsequently leads to a 15.8% longer cabin. Accounting for the square root of the area increase a feasible design can be obtained with at least an 11.5% increase in the fuselage dimensions.

Table 4-3 BWB cabin sizing comparison

Parameters	Units	Values	
Aircraft	-	N3-X	EH2 BWB
Inputs			
Business class	-	50	60
Economy class	-	250	354
Number of bays	-	4	4
Cabin sweep angle	deg	70	70
Cabin height	m	2.6	2.6
Results			
Number of closets	-	4	5
Number of galleys	-	6	8
Number of lavatories	-	10	12
Bay width	m	3.66	3.66
Cabin width	m	14.63	14.63
Minimum cabin length	m	8.44	12.95
Maximum cabin length	m	28.53	33.05

Parameters	Units	Values	
Cabin planform area	m ²	270.4	336.5

If the scaled aircraft were to be designed for Jet A-1 the aforementioned increase would suffice. However, in the LH₂ case, there needs to be enough volume available in the underbelly of the aircraft to accommodate enough fuel to meet the design mission requirements.

4.2.2.2 Tank available space

To determine the final scaling the tank available space is examined. Utilising the geometry of the fuselage derived from [111], several scaling factors are applied between +10% and +20%. Considering the shell geometry of the fuselage as well as the spanwise thickness variation several tanks can be fitted. 6 tanks in total occupy more efficiently the available space both volumetrically and gravimetrically. 5 although lighter overall, leave unused space and 7 even though they occupy more space, weigh more overall.

Figure 4-2 shows the available external tank volume and fuel weight against the discussed increase in fuselage dimensions. This first iteration of results is obtained by imposing the maximum dimensions available for each tank according to the fuselage scaling, their positioning and assuming the tank's insulation and wall thickness. The non-linear increasing trend that is observed in the plot is attributed to the fact that as the fuselage depth is increasing linearly, the under-belly available depth increases slightly more than that as the passenger cabin height remains unchanged with the scaling.

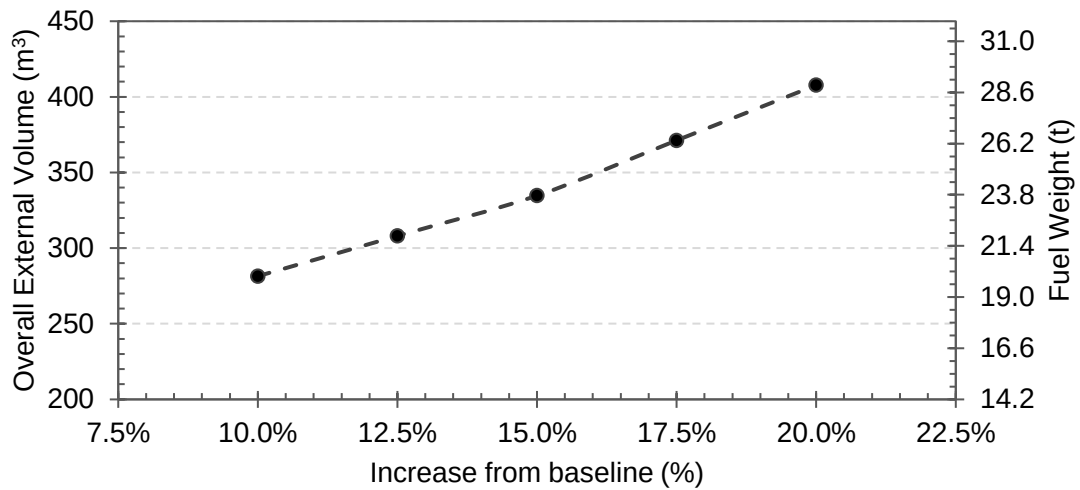


Figure 4-2 Total external tank available volume and approximated fuel weight against fuselage scaling

Through this first approach, it is considered that the +17.5% increase associated with approximately 26.3 t of LH₂ would suffice and the aircraft would meet the range requirements. In this way, the aircraft geometry is fixed and the tank optimisation and propulsion system design space exploration would utilise this geometry.

4.2.2.3 Final geometry

Based on the +17.5% scaling discussed the new dimensions of the aircraft are presented in Figure 4-3. The aircraft length has increased to 49.5m maintaining the wingspan of 65m. The passenger cabin, as sized in Table 4-3, covers 66.7% of that length. This number comes in agreement with the BWB sizing guidelines since after the passenger cabin wall the main spar of the wing is crossing at approximately 70% of the chord of the fuselage [32, 112].

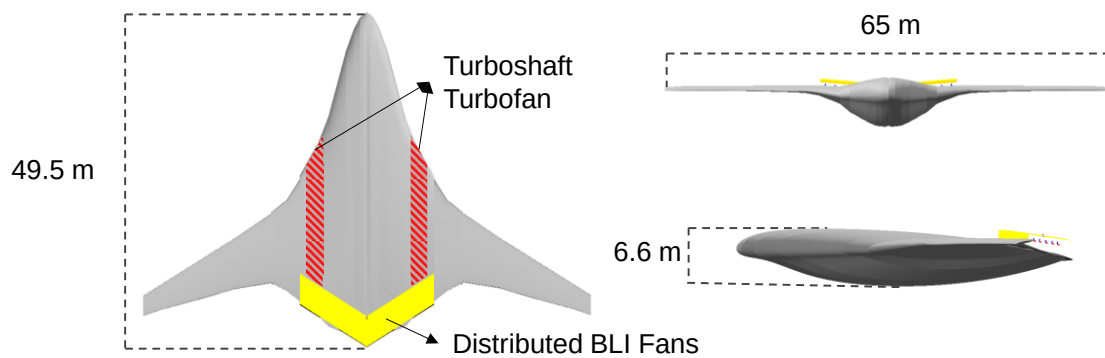


Figure 4-3 Scaled aircraft dimensions

In detail, the characteristics of the airframe components that will be used as inputs both in the aircraft performance as well as the aircraft weight estimation tool, are summarised in Table 4-4. Fuselage section is determined by the passenger cabin width. The wing box is comprised of the outer wing area which has remained unchanged as well as the wing root which has been adjusted so that it smoothly connects the scaled fuselage in a way that the sweep angle of the wing remains constant.

Table 4-4 Sized aircraft components geometry

Parameter	Units	Values			
Component	-	Wing	Fuselage	Eq. Platform	Winglet
Area	m ²	656	556	1212	5.7
Span	m	50.4	14.6	65	3
Aspect Ratio	-	3.93	0.37	-	-
Equivalent Aspect Ratio	-	-	-	4.01	-
Sweep angle	Deg	30.5	70	-	30
Average t/c	-	0.09	0.12	-	0.1

Finally, a side-to-side comparison of the baseline and scaled versions are shown in Figure 4-4.

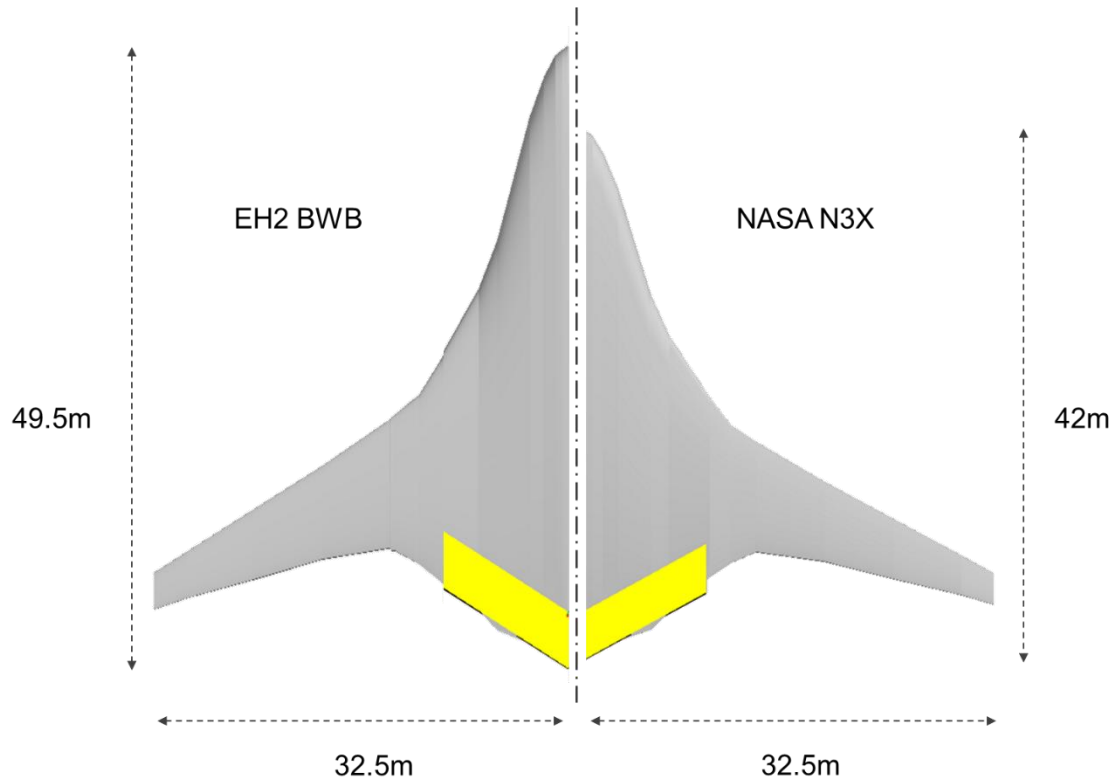


Figure 4-4 Scaled and baseline aircraft geometry comparison

As the geometry and the available volume for the tanks have been identified an initial sizing of the aircraft was performed to establish representative fuel flow profiles throughout the flight and thrust requirements. The former will be used for the detailed sizing of the tanks and the latter for the design space exploration. An initial propulsion system with 10 BLI fans is considered and several iterations between the propulsion system sizing, the aircraft weight estimation and the aircraft performance modules are performed. The tank weight is calculated based on a fixed η_{tank} .

4.2.3 Aircraft weight estimation additional inputs:

For the weight estimation of the aircraft both the cabin sizing presented in Table 4-3 and the geometry of the components in Table 4-4 is used. In addition to these, in order to account for composite structures as in [45], the wings are assumed to have a composite utilisation factor, as introduced by FLOPS [105], of 0.3 (0.0 no composites and 1.0 if only composite structures are used). The wings are also assumed to carry approximately 70% of the aircraft load with the

rest provided by the fuselage section. The combination of these two numbers was established in the aircraft weight estimation validation for the N3A presented in Table 3-2 and allowed for adequate data matching. Finally, the effect of mounting the propulsion system at the fuselage section is also taken into account

4.2.4 Design mission definition

Once the aircraft geometry has been determined, next is the flight profile definition for the design mission. Based on a typical long range flight profile the inputs in terms of altitude and flight speed are presented in Table 4-5. The take-off segment ends at 1500 ft where the target calculated airspeed is 194 knots at a deviation from ISA conditions of +15 K. For the climb and descent segments, a target CAS for the set altitudes are determined and once these have been achieved, they are considered constant for the remaining of the segment. Only for the climb segment, a +10 K deviation from ISA conditions is set.

Table 4-5 Fight profile: altitude and velocities

Parameter	Flight segment	Altitude	CAS	ΔT_{ISA}
Units	-	ft	kts	K
Values	Take-off	1500	194	+15 / 0
	Climb / Descent	3000	250	+10 / 0
		10000	310	+10 / 0
		18000	310	+10 / 0

For the top of climb/descent, a target Mach number is set that would be in the vicinity of the cruise Mach number with a +10 K deviation from ISA. The cruise Mach number is set to 0.83 (long-range cruise) and the altitude to 40000ft. The selection of these is detailed in section B.1 of the appendix.

Apart from the flight profile requirements, the reserves are taken into account as follows:

- Contingency: 5% of trip fuel
- Diversion: 200 nmi at 22000 ft with Mach Number 0.68
- Hold: 30 minutes at an altitude of 1500ft

The aerodynamic performance at the start of cruise of the EH2 BWB together with that of the SAX – 40 [42], (which is considered to be similar of that of the N3-X) is presented in Table 4-6. Although the two aircraft were designed for different applications (215 pax for 4500 nmi for the SAX-40 against 414 pax and 7500 nmi for the EH2 BWB) and with different modelling assumptions, both L/D and C_L characteristics appear to be in the same range. The hydrogen powered EH2 shows a lower L/D since for the same wingspan, the fuselage was designed bigger to accommodate a bigger passenger cabin as well as the hydrogen tanks while the wing section maintained the same span.

Table 4-6 Key aerodynamic characteristics of the EH2 BWB and the SAX-40, data obtained from [42]

Aircraft	SAX – 40	EH2 BWB
Altitude (1000 ft)	40	40
Wingspan (m)	67.5	65
Fuselage length (m)	44	49.5
Cruise	0.8	0.83
L/D	25.2	22.3
C_L	0.2064	0.2474

4.3 LH₂ Tanks

In this section, the LH₂ tanks are optimised to provide the lightest overall weight. Through the initial assessment shown in Figure 4-2, given the available volume of the fuselage, 26.4 t of LH₂ were estimated based on an initial insulation and wall thickness. Using the methodology outlined in sub-section 3.2, the tanks are re-sized to reflect the optimised results.

At first the geometry restrictions, in terms of maximum diameter and length, are established by examining the available volume. The 6 tank layout and its numbering are schematically shown in Figure 4-5.

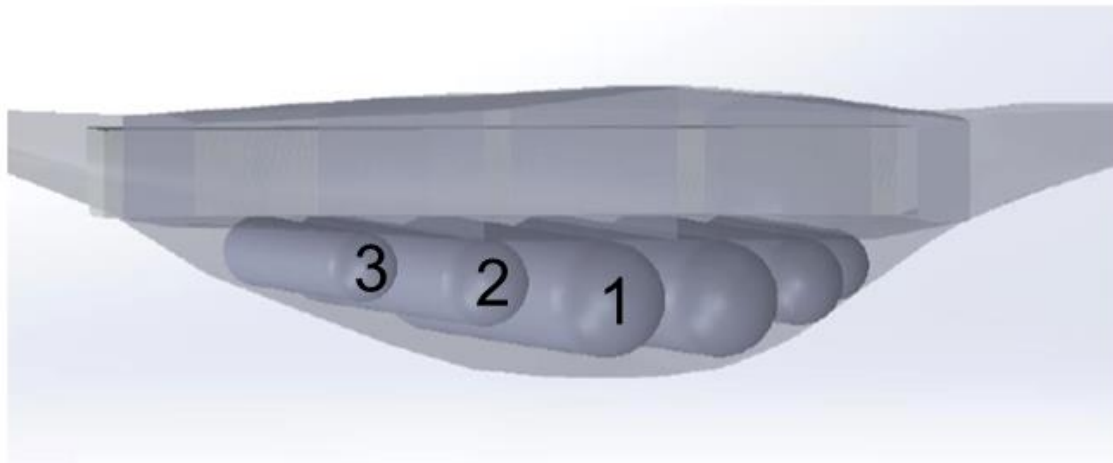


Figure 4-5 Tank numbering and positioning in the underbelly volume of the fuselage

Based on the fuselage dimensions the maximum available diameter and length for every tank are presented in Table 4-7. Tank number 1 has the biggest dimensions as it is closest to the centreline of the aircraft, where the fuselage depth is bigger. For tanks 2 and 3 as the depth reduces both the maximum diameter and length are restricted to lower values.

Table 4-7 Sized aircraft components geometry

Parameter	Units	Values		
Tank numbering	-	1	2	3
Maximum external diameter	m	2.82	2.025	1.63
Maximum total length	m	24.38	22.71	15.55

To size the tanks, the maximum external diameters for each one are set as inputs. Al 2219 is used for the wall and its properties together with other wall sizing parameters are taken from Table 3-5. Based on the insulation material assessment presented in Figure A-2, polyvinylchloride closed rigid foam cell is used for the insulation [96].

The next step is to determine the fuel split between the tanks. Starting with the initial estimate of 26.4 t of LH₂ a parametric sweep is performed where for each tank the fraction LH₂ stored and the insulation thickness is varied. For each insulation thickness, there is a corresponding fuel fraction that provides the maximum allowable external length. Additional considerations include:

1. The engine fuel flow is taken from an initial assessment of the propulsion system discussed in 4.2.2.3 and is considered that it will not vary significantly to influence the tank sizing when performing the design space exploration for the propulsion system.
2. Hydrogen for engine use is extracted simultaneously from all tanks and the corresponding flow rates have the same fraction as the tank stored to total fuel fraction.
3. At the end of the mission, all tanks have reserve fuel equal to the corresponding stored fuel fraction of the total reserves required, approximately 2600 kg.
4. A two-hour ground hold with an ambient temperature deviation from ISA of +15 K is considered.
5. Initial fuel pressure is 1.4 bar and the volume of GH₂ occupies 3% of the total volume, as per the example of sub-section 3.3.4.
6. It is assumed that during taxi, take-off, climb, descent and landing, the tank contents are fully mixed.

Two cases are examined, in the first the tanks are designed to withstand the pressure build-up throughout the flight without the need to extract GH₂ for pressure control. For the second case, GH₂ is extracted during cruise at a rate that maintains level the pressure and is considered to be used in the engines together with LH₂.

The parametric sweep for the two cases is presented in Figure 4-6. As outlined in the example of section 3.3.4.3, the tanks that feature GH₂ extraction for pressure control, allow for a superior η_{tank} which suggests lighters designs. Between the two cases, for the different tank optima, GH₂ extraction tanks achieve the lightest designs at a lower insulation thickness. This allows for greater internal volumes and more LH₂ storage. Last but not least, between the different tanks a bigger η_{tank} is obtained with bigger external diameter and volume.

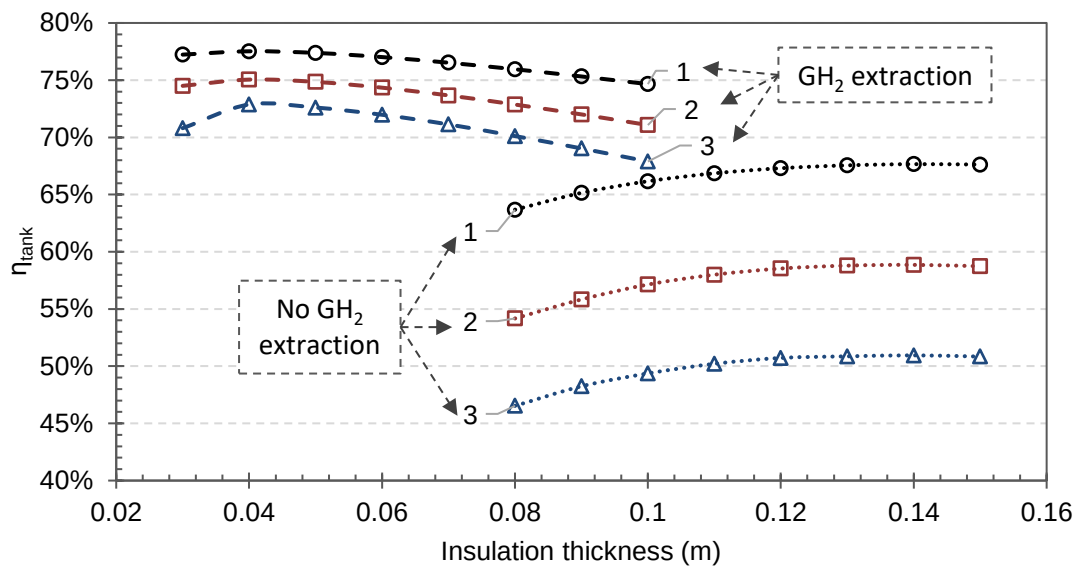


Figure 4-6 Tank gravimetric efficiency against insulation thickness for all the tanks with and without GH₂ extraction during cruise

A more detailed insight can be obtained by examining the differences in the designs with the maximum η_{tank} between the two cases presented in Table 4-8. First the lower insulation thickness allows for greater internal volume in the order of +15% for tank number 1 and up to +26% for tank number 3. The significant increase in internal volume results in an increase in both total and useful LH₂ stored which overall accounts for +18%. The lower insulation thickness provides for lighter insulation in all cases. The lower pressure levels due to GH₂ extraction allow for lighter walls. In total, a significant weight saving of 40% is achieved. However, since GH₂ is partly used for the engine fuel flow demand and due to the higher heat ingress of the tank at least 2.5 times more

LH₂ is allowed to evaporate. Between all tanks, bigger benefits are obtained with smaller external dimensions. More details about the sizing of these designs are presented in Table B-1 and Table B-2 of the appendix.

Table 4-8 Differences between the maximum η_{tank} designs for the GH₂ and no GH₂ extraction cases

Parameter	Difference between GH ₂ extraction and no GH ₂ extraction cases		
Tank numbering	1	2	3
Internal Volume	14.9%	20.8%	25.9%
Total weight	-32.0%	-44.3%	-51.8%
Useful LH ₂	14.5%	20.8%	28.0%
Gravimetric efficiency	13.7%	26.6%	42.7%
Mission evaporated LH ₂	235.9%	295.9%	333.7%
Maximum allowable pressure	-18.4%	-40.0%	-52.9%
Insulation thickness	-67.9%	-67.9%	-67.9%
Total LH ₂ stored	17.7%		
Total useful fuel	17.7%		
Total tank weight	-39.5%		

In Table 4-9, for the discussed cases above, the ratio of unusable LH₂ to the useful quantity stored is presented. For the GH₂ extraction cases that ratio remains approximately constant at 3%, whereas for the tank designs with no GH₂ extraction, this increases with decreasing tank size.

Table 4-9 Unusable LH₂ to useful LH₂ stored for GH₂ and no GH₂ extraction

Parameter	Unusable LH ₂ / Useful LH ₂		
Tank numbering	1	2	3
No GH ₂ extraction	3.1%	3.8%	4.7%
GH ₂ extraction	2.8%	3.0%	3.1%

Although the GH₂ extraction strategy allows for greater benefits in weight savings and fuel stored, examining the pressure evolution for each tank raises the issue of handling after the mission. More precisely the gauge pressure

evolution for the three GH₂ extracting tanks is presented in Figure 4-7. The smaller the tank the bigger the pressure rise during ground hold, although in none of the three cases does the pressure exceed the maximum. At the end of the mission, tanks 1 and 2 have a slight pressure margin compared to the maximum, which would allow for a very short period without the need for pressure control. Compared with the start of the mission, in both cases, no engine fuel flow is withdrawn from the tanks but at the end the tank is almost empty of LH₂, which would suggest that the pressure will build up approximately 4 times faster than that at ground hold (see example of evolution of Φ in plot (c) of Figure A-1). Pressure control mechanisms should be considered such as:

1. Tank content mixing.
2. GH₂ extraction to power the auxiliary power unit or in worst case vent overboard the aircraft.
3. Immediate connection of the tank with the LH₂ and GH₂ supply lines of the airport.
4. GH₂ extraction scheduling for engine use during the descent that will allow greater pressure drop.

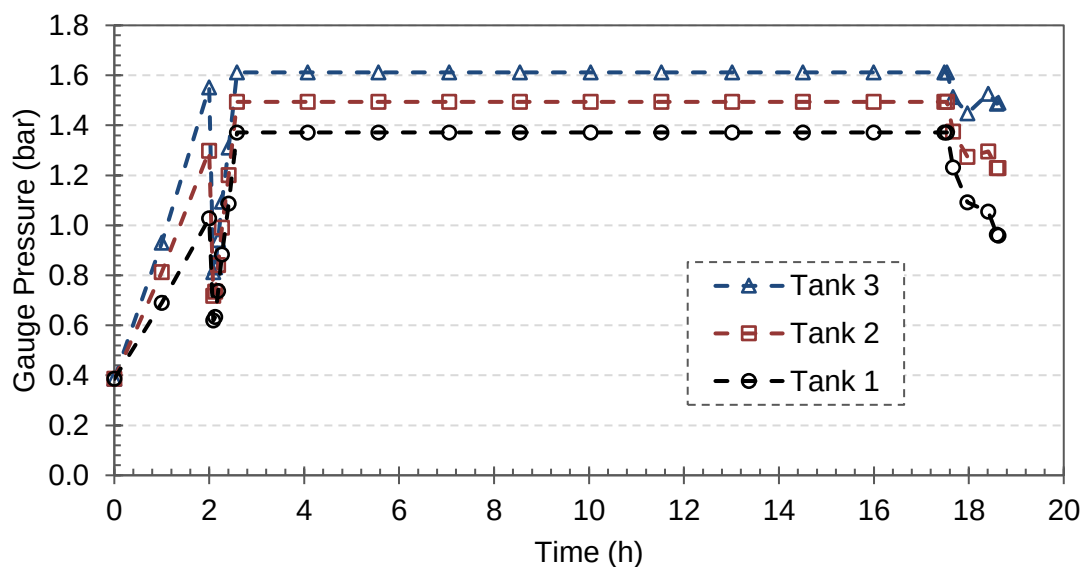


Figure 4-7 Gauge pressure evolution for the three tanks against the mission time

Although tanks 1 and 2 provide a small pressure margin, tank number 3 does not. A combination of rising ambient temperature during descent, relatively low

engine fuel flow demand and the tank dimensions lead eventually to a small rise in pressure.

To avoid this an alternative sizing strategy is adopted which involves:

- Tanks 1 and 2 to be sized to carry all the trip fuel and by the end of the mission are almost empty of LH₂. The rest of the sizing considerations remain the same as discussed above.
- Tanks number 3 are sized to carry all the reserve fuel, approximately 2.6t. An additional 12 hours after the flight are included for the sizing and no LH₂ or GH₂ are extracted throughout the whole period. In that way, if a quick turnaround or a long stay after the flight are required, pressure build-up in the tank could be allowed. As the sizing requirements have changed for tanks 3, the 3% volumetric ullage of GH₂ at the beginning of the flight is increased to 8% to accommodate the LH₂ expansion throughout that period. Constant fuel mixing is applied to minimise the pressure build-up.

The optimised results for the three tanks are presented in Table 4-10. Tanks 1 and 2 have not significantly changed, they have a relatively high η_{tank} and carry approximately 62% and 29% of the on-board fuel respectively. Although they allow for at least 10% of the initially stored LH₂ to be evaporated only 3% is considered unusable for both cases. For tank number 3 since no fuel is withdrawn from these tanks, no LH₂ is evaporated and the tanks do not have to be oversized to compensate for the unusable fuel. This overall results in a η_{tank} of 56.4% which is higher than the no GH₂ extraction sizing strategy presented in Figure 4-6.

Table 4-10 Tank sizing results

Parameter	Units	Values		
Tank numbering	-	1	2	3
External diameter	m	2.82	2.025	1.63
Total length	m	24.23	22.57	15.21
Length limit	m	24.38	22.65	15.54
Internal Volume	m ³	135.3	64.2	20.2
Total weight	kg	2303	1262	997
Useful LH ₂	kg	8779	4153	1290
Mission vented GH ₂	-	601	439	0
Mission evaporated LH ₂	kg	892	589	0
Unusable LH ₂	kg	292	150	0
Maximum allowable pressure	bar	1.41	1.54	2.46
Insulation thickness	m	0.04	0.04	0.144
Fuel split	-	61.7%	29.2%	9.1%
Tank weight split	-	50.5%	27.7%	21.9%
Gravimetric efficiency	-	77.2%	74.6%	56.4%
Mission vented GH ₂	-	6.6%	10.2%	-
Mission evaporated LH ₂	-	9.8%	13.7%	-
Unusable LH ₂	-	3.2%	3.5%	-
Total LH ₂ stored	t	29.32		
Total useful fuel	t	28.44		
Total tank weight	t	9.12		

The overall fuel stored is 29.3 t, however, only 28.44 t can be considered usable by the engines. The latter number will be used for the aircraft performance that will follow. For the weight estimation of the aircraft both the total tank weight of 9.12 t and the weight of the unusable fuel stored of 0.9 t will be accounted for. The pressure evolution of the three tanks throughout the design mission is presented in Figure B-2 of the appendix.

4.4 Propulsion system inputs and subsystem performance

Before addressing the integrated propulsion system design space exploration, the inputs and constraints for the propulsion system sizing are discussed. In addition, to better explain the propulsion system results in the subsequent sections, the performance of some subsystems is first analysed hereby.

4.4.1 Propulsion system inputs

Multipoint design conditions and thrust requirements

Two operating points are considered for the multi-point sizing, top of climb (ToC) as the design point (DP) and end of runway (EoR) as off-design (OD). During the former, the gas turbine generally operates at its highest non-dimensional power setting and at the latter maximum component operating temperatures, pressures and mass flows are observed. The ambient conditions and thrust requirements that will be used as inputs for the entirety of the propulsion system design space exploration are summarised in Table 4-11. The DP conditions are set to represent a high power setting that would be representative of ToC for various mission profiles. The thrust levels selected, were derived from the initial assessment considering representative MTOW requirements, adequate climb rates as well as overall decent time. Ideally, these would need to change throughout the design space as the weight of the propulsion system changes. However, the overall difference in MTOW would only impose minor corrections to the thrust requirements that will not significantly influence the final results and given the added complexity in the sizing routine, this aspect is not further addressed.

Table 4-11 Multipoint operating conditions and requirements

Parameters	Units	Values	
Operating Condition	-	DP (~ToC)	OD (EoR)
Altitude	ft	35000	0
Flight Mach Number	-	0.82	0.25
dT_{ISA}	K	+10	+15
Net Thrust	kN	72.6	240

Design Point inputs

For the design point calculation the required inputs are summarised in Table 4-12. The component isentropic efficiencies are selected considering ambitious projections for 2050 from previous studies, [28, 55, 146, 147]. The fan efficiency in particular is targeted at 95%. However, in order to account for the inefficiencies of a BLI fan, due to the distorted flow at the inlet, a penalty of 1.5% is applied as adopted in [28] (although recent studies [148, 149] that through the optimisation of the fan stage design this can further reduce to 1%). For the inlet recovery of a semi-buried turboshaft installation, 0.5% loss is introduced, which is slightly less than value assumed in [146] for a podded application. 3% of heat exchanger hot stream and exhaust duct losses combined are applied as discussed by Brewer [20]. For the NGV's, assuming that ceramic matrix composite materials will be used, cooling requirements will be reduced and hence a fixed 6% bleed air from the HPC exit is used for the HPT and an additional 1% from the IPC exit for the IPT. For all the shafts, mechanical losses of 0.5% are considered. The heat exchanger effectiveness is set at 0.7 at the design point. Finally, for the calculation of the fuel injection properties, the fuel injection pressure is set constant to 60 bar, as it needs to be higher than the absolute pressure of the HPC exhaust and the LH₂ stored temperature is set constant to 21 K. To account partly for the power requirements of the fuel system as well as for cabin use, a power off-take of 250 kW is assumed and is applied throughout the whole flight.

Table 4-12 DP performance inputs

Parameters	Units	Values
Isentropic Efficiencies		
IPC	-	91.5%
HPC	-	91%
HPT	-	91%
IPT	-	92%
LPT	-	94%
FAN _{BLI}	-	93.5%

Parameters	Units	Values
Pressure losses		
Turboshaft Inlet recovery	-	99.5%
IPC to HPC	-	1%
Combustor pressure losses	-	4%
Heat exchanger hot stream losses	-	3%
BLI intake additional losses	-	0.5%
Miscellaneous		
Cooling bleed: HPT from HPC exit	-	6%
Cooling bleed: IPT from IPC exit	-	1%
Shaft Mechanical Losses	-	0.5%
Combustor Efficiency	-	99.9%
Power Off-take from the generator	kW	250
Initial Fuel Temperature	K	21
Fuel Injection Pressure	bar	60

Turbomachinery gas path sizing and weight estimation

For the turbomachinery weight estimation, inputs for all the components are set and are presented in Table 4-13. A combination of axial Mach numbers, hub to tip ratios as well as mean line evolution of all the components are set across the gas path to enable compact designs. This sizing approach adopted aimed at reducing as much as possible the tip diameter of the components in an effort to maximise their rotational speeds. This first-order simplification allows for a universal sizing throughout the design space, however, may not necessarily lead to an optimum sizing all the time.

For the BLI fans, their small size would potentially allow for a hub to tip ratio of 0.27. Additionally, a constant tip design is selected to allow for fan exit hub diameters big enough to house the HTS motor powering the fan. For the components of the turboshaft rather radical hub to tip ratios are selected. At the inlet of the IPC, 0.55 is set to minimise tip diameter at the inlet and maximise the rotational speed that would allow for increased stage loadings. The constant

hub mean-line evolution is selected for a smooth transition of the duct that connects the IPC and the HPC. The size of the HPC is driven by the exit hub to tip ratio, which is selected at a maximum allowable value of 0.92 to avoid small stator blade heights at the exit [138]. In reverse, the hub tip ratios for the turbines are set. For the LPT in particular an exit hub to tip ratio of 0.5, as suggested in [142], is set to maximise the rotational speed of LPT and in turn allow for lighter LPT, generator and motor designs.

The number of stages and the axial Mach numbers are set as direct inputs for the gas path sizing and weight estimation for all the components apart from the LPT. They are selected after a few iterations considering stage loadings, resulting component efficiencies and reasonable geometry of the gas path, using the ATLAS. For the LPT, the number of stages is left unconstrained as its power requirement would significantly vary throughout the design space and therefore the stage loading per stage.

Table 4-13 Turbomachinery geometry inputs

	Component		Mn_{Axial}	Hub Tip Ratio	Mean line Evolution	Number of Stages
Turboshaft	BLI Fan	In	0.68	0.27	Constant Tip	1
		Out	0.54	-		
	IPC	In	0.6	0.55	Constant Hub	4
		Out	0.5	-		
	HPC	In	0.45	-	Constant Hub	9
		Out	0.27	0.92		
	HPT	In	0.1	0.9	Constant Tip	2
		Out	0.4	-		
	IPT	In	0.4	-	Constant Tip	1
		Out	0.45	-		
	LPT	In	0.45	-	Constant Hub	TBD
		Out	0.5	0.5		

Sizing restrictions

In addition to the geometry inputs discussed, additional restrictions are imposed. First, the maximum allowable TET and HPC exit temperature, T₃, are set to 1850K and 1030K respectively. These are met at end of runway

conditions and are the main driver for the selection of TET and OPR at the DP. In addition to the T3 limit, the blade height at the exit of the HPC is also checked. However, a combination of core mass flow (associated with the application), axial Mach number and hub to tip ratio selected, prevented small blade heights throughout the design space. For the rotational speed of the fans and the compressors two limits are set, maximum tip blade speed and maximum tip relative Mach number. For big diameter fans, the tip speed is likely to restrict the rotational speed and for ones with smaller diameters the relative Mach number at the tip. In this case, a maximum rotational speed of 450 m/s and a relative Mach number of 1.6 at the blade tip is imposed [138]. For compressors, the equivalent restrictions selected are 400 m/s and 1.3 respectively. For the HP and IP turbines, a maximum blade tip speed of 480 m/s is selected to ensure rim speeds below 400 m/s. For the LP turbine, the same limit is imposed (while rim speed is below 350 m/s) but a limit on the AN^2 of 50 m^2rpm^2 is also included. All the speed limits discussed are based on [138, 142]. The shaft speeds are then chosen considering all the limits of the connected components. Finally, and similarly to [28], the maximum array width is set to 20 m, as the width of the fuselage between the N3-X and this application has remained unchanged. A wider array, extending towards the wings, would potentially benefit the performance of the propulsion system as more boundary layer is ingested. However, this comes in conflict with the requirement of large control surfaces since the N3-X is a tailless configuration.

Table 4-14 Sizing restrictions

Parameters	Units	Values
Maximum TET	K	1850
Maximum T3	K	1030
Fan Maximum Tip blade speed	m/s	450
Fan Maximum tip relative Mach number	-	1.6
Compressors Maximum Tip blade speed	m/s	400
Compressor Maximum tip relative Mach number	-	1.3
Turbine Maximum Tip blade speed	m/s	480

Parameters	Units	Values
LPT maximum AN^2	m^2rpm^2	50
Maximum fan array width	m	20

BLI fan array installation

The nacelle of the BLI fans is assumed to be a flat plate with width (for spacing between propulsors) and length 1.15 and 2 times the fan diameter respectively. The nacelle dimensions are then used to estimate the nacelle drag of the array and its weight is estimated as in [142]. For the turboshaft, no nacelle installation is considered in the calculation due to the semi-buried configuration.

Transmission ratio

As discussed in section 3.4.6.3, the list of pole pairs for both motors and the generator are presented in Table 4-15. This represents the potential transmission ratio between the LPT and the remote fans, it can only take discrete values as it depends on the pole pair ratio. The maximum number of poles pairs (and therefore a denser selection of transmission ratio) is restricted to 5 to avoid high AC losses (due to higher frequencies) and added weight on the HTS machines.

Table 4-15 HTS potential pole pair selection and resulting transmission ratio

Generator pole pairs	1	1	2	3	4	3	5	4
Motor pole pairs	3	2	3	4	5	3	4	3
LPT to BLI FAN transmission ratio	0.33	0.50	0.67	0.75	0.80	1.00	1.25	1.33

Electrical component sizing and performance

For the weight estimation of the HTS motors and generators the rated power and rotational speed is required, as derived from the multi-point design routine. The maximum power is met at EoR and the maximum rotational speed at ToC and so this combination is used to size the components to ensure that the HTS

performance does not exceed the limits discussed in section 3.4.2.1. For the sizing apart from the rated power and rotational speed as well as pole pair that vary, the inputs in Table 3-8 are used.

The HTS efficiency is estimated based on the rated requirements. For the design space examined and based on the assessment discussed in section 3.4.2.1 as well as the efficiency estimation of Figure 3-27, the efficiency is expected not to vary significantly from 99.45% for generators and 99.35% for motors. Accounting for the reduced power relative to the rated for ToC, 99.4% and 99.3% are set fixed for all the DP calculations.

4.4.2 BLI Propulsor performance

In this subsection, the performance of a BLI fan is discussed. Previous studies [27, 59, 78, 122] have showcased how the BLI propulsor sizing and installation parameters affect its performance. However, to better understand the overall performance of the propulsion system, a similar assessment is presented herein.

For the calculation of the BL velocity profile, the 7th-power law is chosen similarly to [27, 77, 79]. This provides a better matching with the flow characteristics predicted with CFD over the N2-A NASA BWB model presented in [122]. The N3-X effectively has the same fuselage characteristics as the N2A and the same boundary layer characteristics have been used in related studies. Last but not least, the local Mach number, apart from the free-stream Mach number, also depends on the position of the frame of reference from the leading edge. According to [122], at 35000 ft and for a flight Mach number of 0.8, the local Mach number above the fuselage at 85% of the centreline chord at the of the aircraft is lower by 2.5%. These conditions are considered representative of ToC and cruise conditions but since extensive data over the whole flight envelope have not been acquired, the local Mach number is going to be considered 2.5% lower than the flight Mach number throughout the mission as in [79, 122].

With the considerations above, an example is presented below where the sizing of a BLI fan installed above the BWB fuselage is examined, based on the ToC ambient conditions of Table 4-11 and the related performance inputs of Table 4-12. For the example three parameters are varied:

1. The net propulsive force of the fan, NPF
2. The fan pressure ratio, FPR_{BLI}
3. The fan spanwise position over the fuselage that imposes a variance in the length of chord, l_{chord} and hence the BL thickness

Figure 4-8, shows the dependency of mass flow requirement, (a), and control volume width, (b), against FPR_{BLI} over three NPF levels. Expectedly, the lower the FPR_{BLI} and the higher the thrust requirement, the higher the mass flow requirement. As the control volume width is assumed to be the same as the diameter of the fan and since for all the parameters the fan is sized based on the same fan face Mach number and hub to tip ratio, its diameter will follow the same trend as the corresponding mass flow.

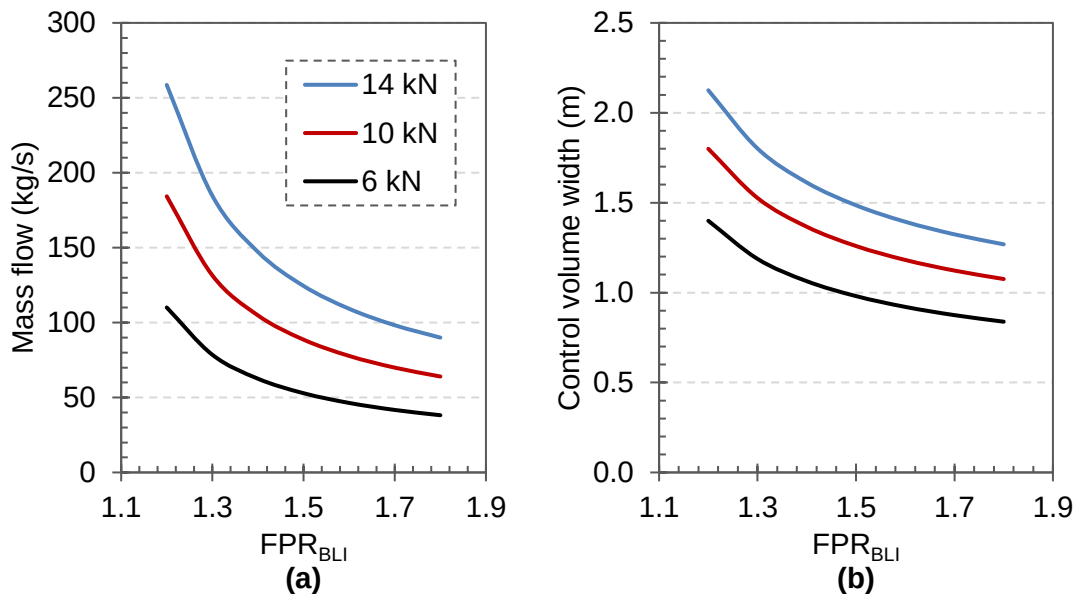


Figure 4-8 Mass flow requirement (a) and control volume width (b) over FPR_{BLI}

The inlet flow properties and power requirements are summarised in Figure 4-9. In the top figure, the average total pressure and velocity loss are estimated depending on the proportion of the boundary layer ingested, h/δ . In the bottom figure, the power consumption is mapped for the NPF and FPR_{BLI} requirements

for three different lengths from the leading edge (20, 30 and 40m) over the same parameter. The power consumption in principal follows the trend of the mass flow but shows variance depending on the propulsor positioning. In more detail, between the 40m and 20m positioning for the 6 kN thrust output, the latter requires approximately 2 – 3% more power for the same FPR_{BLI} . For the 14 kN case, since more free stream air is ingested (higher h/δ) the boundary layer effects are moderated and the 20m case requires only 1.5 – 2% more power. The 20m case roughly represents the most outward propulsor and the 40m the one closest to the centreline of the fuselage. The differences in BLI properties and power consumption depending on the sizing requirements signify the importance of accounting for the different number of fans as well as their positioning.

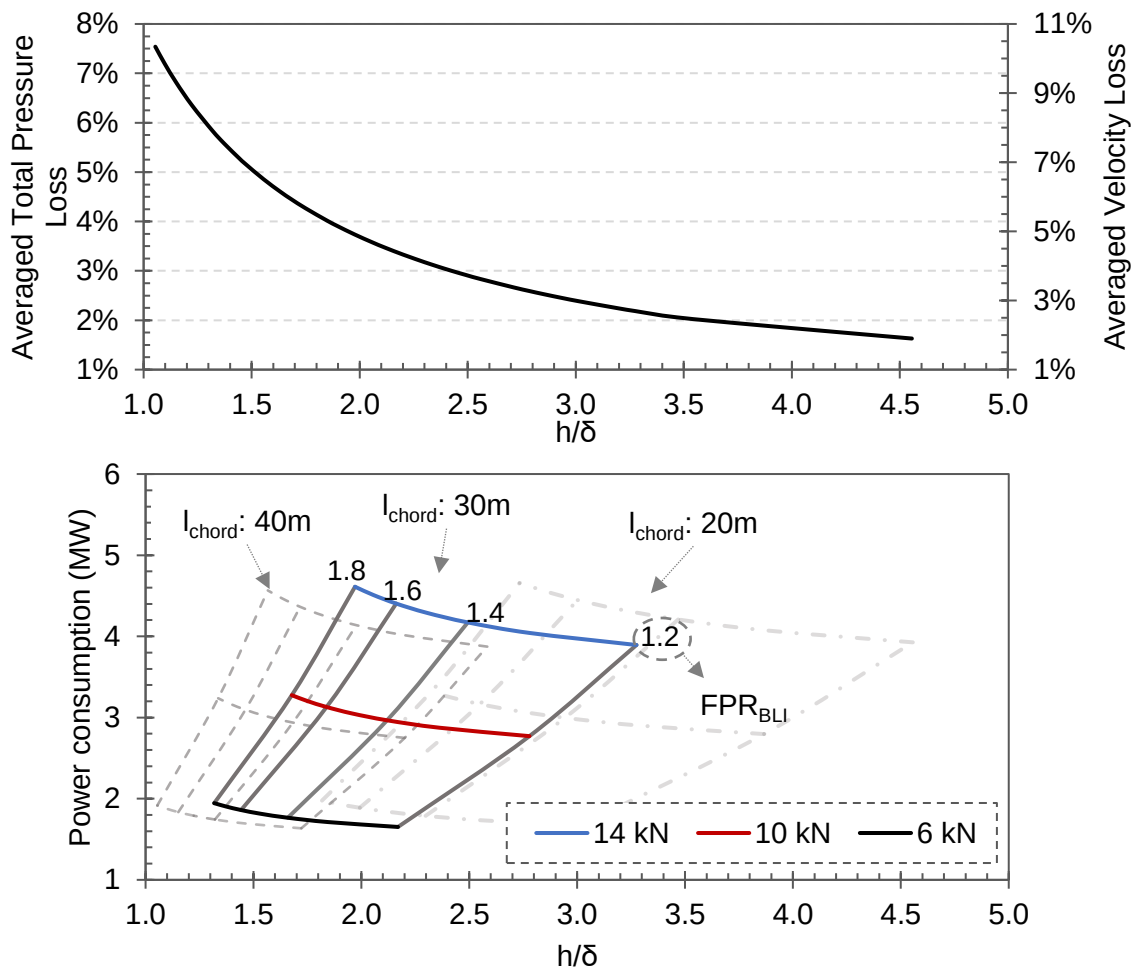


Figure 4-9 (Up) averaged total pressure and velocity loss, (down) Power consumption for indicative NPF and FPR_{BLI} over h/δ

Section B.3 of the appendix includes a comparison of the BLI performance results discussed above, with equivalent-sized propulsors operating in free stream flow.

4.4.3 Thrust split between hot (core flow) and cold (bypass or BLI flow) gas paths

One common aspect of every turbofan sizing is the BPR optimisation for a given FPR and vice versa. As suggested in [51, 150], for every BPR there is a FPR that minimises the engine fuel consumption. Similarly to the turbofan in the distributed propulsion application, the bypass ratio of the system (BPR_{sys}) can be introduced that describes the ratio of the mass flows producing cold to hot thrust. However, the BPR_{sys} only provides information about the overall system and not the individual BLI fan performance. Therefore to avoid the complexity and added computational time associated with an optimisation process, as well as increase the robustness of the DP calculation, another approach is followed. Instead of using the BPR_{sys} as a design parameter at the DP, the thrust split between the core flow and the BLI fans (equivalent to bypass flow of a turbofan) is chosen instead. The BPR_{sys} then becomes an output to the calculation.

To better discuss the thrust split choice an example is considered, where a single turboshaft engine is powering 6 BLI fans (12 Fan array in total). At DP using the inputs discussed in Table 4-11 and Table 4-12, for an OPR of 85 and a TET of 1800K, the FPR of the fans is varied together with the thrust split. It is reminded that all the fans are sized identically and therefore the fan closest to the centreline operates at the design FPR established while the rest in slightly off-design conditions. The resulting SFC values are presented in Figure 4-10 against the thrust percentage produced by the core for various FPR_{BLI} values.

It is observed that a value of 1.3 allows for the lowest SFCs and therefore more efficient systems. FPR of 1.2 was also examined but provided a trend similar to the 1.4 FPR and is not shown in the graph. More importantly though, for every FPR there is a thrust ratio that allows for the lowest SFC and it does not vary

significantly. For low FPR 1.3 – 1.5 this is 5% and can go up to approximately 6.5% for 1.8.

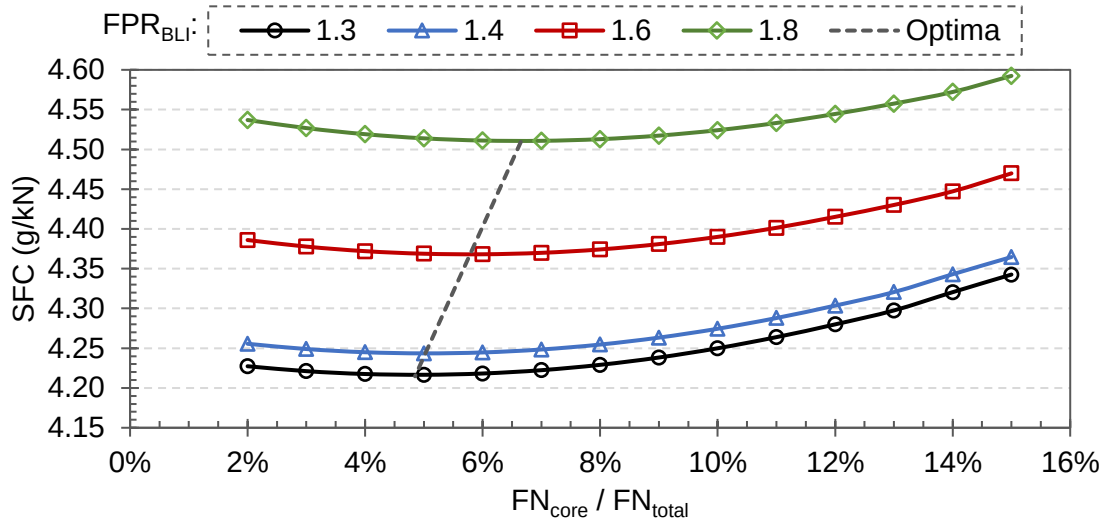


Figure 4-10 SFC against thrust split for different FPR_{BLI} , OPR 85 & TET 1800 K

As the OPRs and TETs will vary throughout the design space a few more sets are considered to check whether the optimum thrust split varies. Figure 4-11 presents the evolution of SFC against FPR for two additional sets of TET and OPR of 1750K & 85 and 1800 & 90 respectively. Similarly as before the FPR of 1.3 shows the lowest SFC across constant TET and OPR, which is a trend that agrees with previous studies [55]. At the same time the optimum core thrust ratio, as also indicated in Figure 4-11, does not vary with different TETs and OPRs.

Finally, the BPR_{sys} for the optima thrust splits against the FPR for the same example is presented in Figure 4-12. Notably, the BPR decreases with increasing FPR, at the same time higher TETs allow for a greater increase in BPR than higher OPR.

In summary, 5% of core net thrust is considered to be representative of the optimum thrust split regardless of the FPR, OPR and TET and will be used as a direct input of the DP throughout the design space exploration for both turboshaft and turbofan configurations coupled with BLI fans. Although at higher

FPRs 6% would be a more appropriate value to use, the difference in SFC is less than 0.1%.

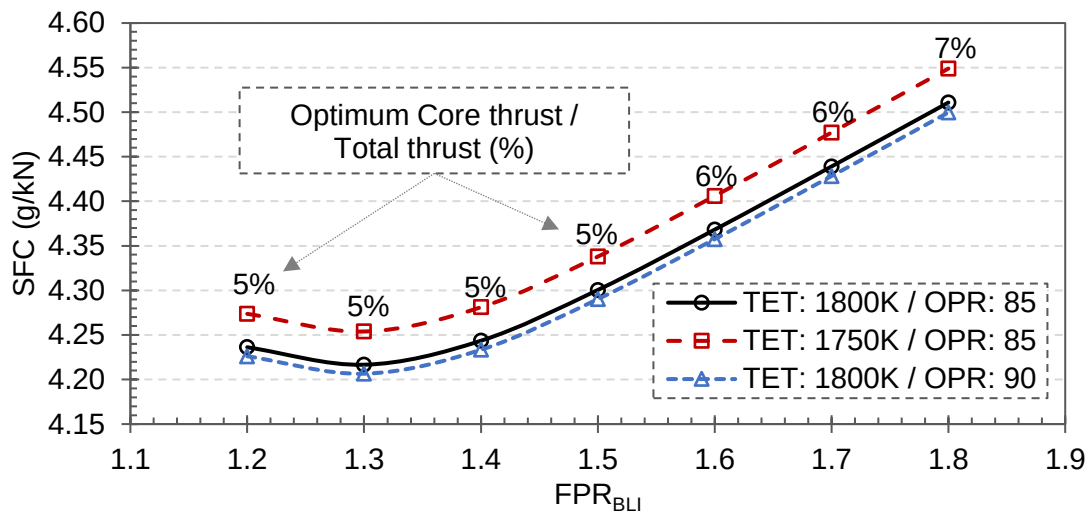


Figure 4-11 SFC against thrust FPR_{BLI}, for different sets of OPR & TET

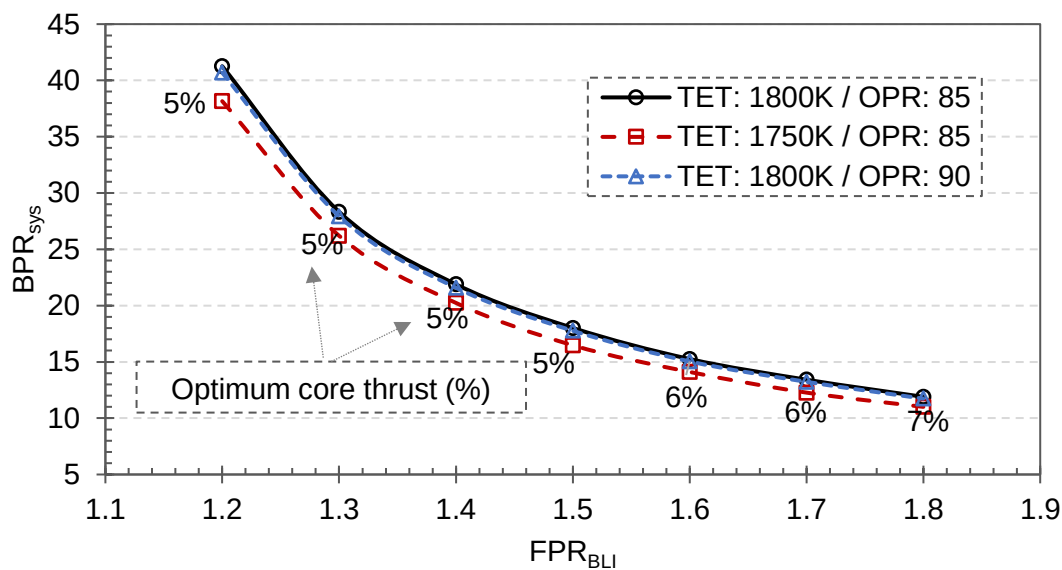


Figure 4-12 Optimum BPR against FPR_{BLI} for different sets of OPR & TET

4.4.4 Fuel properties on engine cycle sizing and injection temperature

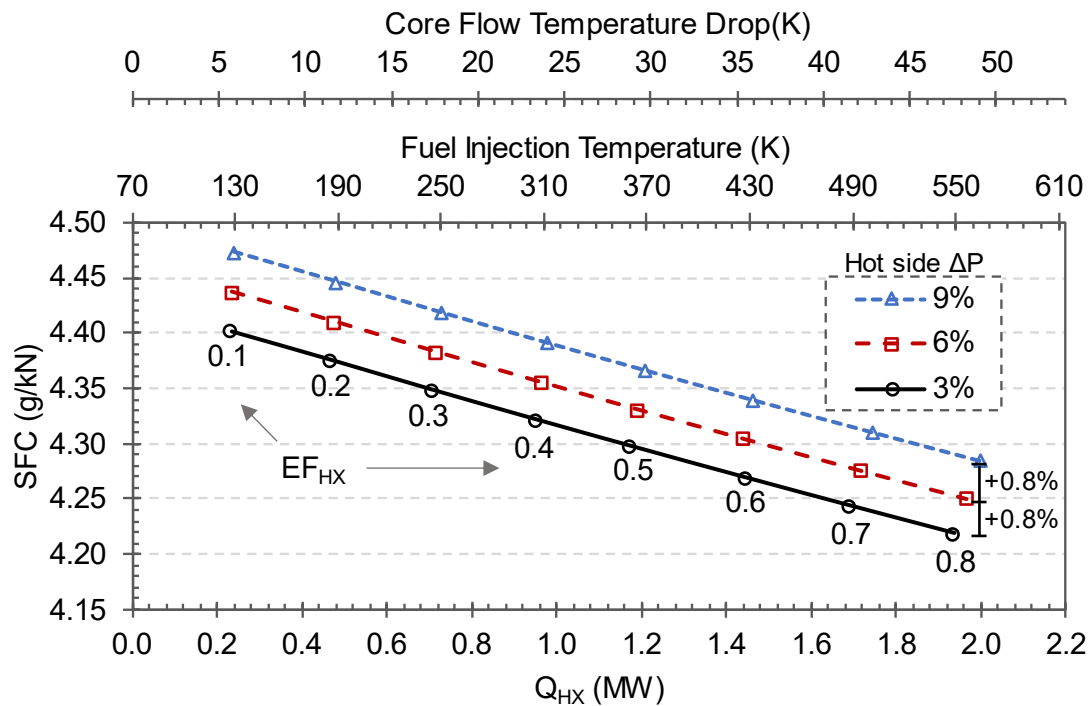
As the fuel is stored at approximately 20K it needs to be heated before combustion to temperatures at least 300-400 higher than that. In this study a

detailed fuel preheating system is outside of the scope of the work, however, the effect of fuel pre-heating utilising energy from the main gas path is addressed as it effectively changes the cycle design. As introduced in Figure 3-34, in a first-order approximation, all the heat losses from the electrical transmission components are assumed to be dissipated to the fuel. Additionally to that, a heat exchanger is installed after the LPT and extracts heat from the main gas path of the core and further heats up the fuel. In this section, the effect of the heat exchanger effectiveness, as well as fuel injection temperature to the combustor, are addressed.

To better understand the effect of the heat exchanger sizing at DP, the example of the previous section is used (Turboshaft & 12 Fans) with the following parameters:

1. TET: 1800K
2. OPR: 85
3. FPR: 1.4

In a parametric sweep, the heat exchanger effectiveness is varied for several values of pressure losses on the hot side. The effect on the SFC against the heat transferred, the fuel injection temperature, as well as hot side temperature drop, are presented in Figure 4-13. The cold side inlet temperature is approximately 80K. For every pressure loss considered, increasing the heat exchanger effectiveness increases the heat transferred between the two sides, the exit temperature of the cold side, i.e. the fuel injection temperature, as well as the temperature drop on the hot side. Interestingly, even though energy is removed from the core flow, which would otherwise translate to a penalty in fuel consumption, the SFC decreases due to the higher heat input in the combustor related to the higher temperature of the fuel. Greater pressure losses on the hot side can be afforded depending on the HX design, as for every 3% increase, the SFC is only penalised by approximately 0.8%.



**Figure 4-13 SFC against heat transferred in the air to fuel heat exchanger for TET
1800K, OPR 85 and FPR_{BLI} 1.4**

From the example, the highest HX effectiveness possible is desired, however, from a sizing point of view, these might not be feasible. On the other hand low effectiveness at DP might result in insufficient fuel heating at other phases of the flight like OD, which will be thoroughly discussed in the next section. Overall an effectiveness of 0.7 at DP with 3% pressure losses on the hot side is selected, similarly to the application proposed by Brewer [20]. These values would be adopted for the entirety of the design space. The weight of the HX is then based on the same design and is scaled according to the heat transferred.

As the HX requirements have been established, a comparison between the DP performance of the same configuration and inputs but with Jet A-1 and hydrogen as fuels is presented in Table 4-16. The fuel injection temperature for Jet A-1 is assumed to be 330, whereas for hydrogen is calculated based on the fuel pre-heating considerations discussed. Starting from the fuel flow (WF), there is an expected difference of -65% compared to Jet A-1 due to the difference in properties which propagates to other fuel depended measures such as the SFC (as the net thrust is the same), the specific power consumption

(SPC) and the fuel to air ratio at the exit of the combustor. Apart from the W_f , as the heat input into the combustor is greater than the Jet A-1 case, the mass flow at the inlet decreases by 6.7% which also reflects to a small increase in BPR. Although the SFC is decreased by 65%, because of the different properties of the fuel, the specific energy consumption (SEC) is a better metric to evaluate the efficiency of the two applications. In terms of SEC a decrease of 3.8% is observed just by switching fuels considering that the two configurations are designed to deliver exactly the same thrust.

Table 4-16 Comparison of Jet A-1 and H₂ as fuels powering a Turboshaft coupled with 6 BLI fans (total array of 12)

Parameter	Units	Values		
Fuel Type		Jet A-1	Hydrogen	dif
Fuel Injection Temperature	K	330	487	47.5%
SFC	g/kN	12.28	4.24	-65.4%
W1	kg/s	38.07	35.54	-6.7%
Fuel to Air Ration at CC exit	-	0.02572	0.00952	-63.0%
W_f	kg/s	0.8917	0.3082	-65.4%
BPR	-	20.4	21.9	7.1%
HX.Q	MW	-	1.69	
SPC	g/MW	40.53	14.01	-65.4%
SEC		1.748	1.681	-3.8%

4.5 Propulsion system design space exploration: Turboshaft-powered TeDP

This section introduces the results of the design space exploration of the turboshaft powering several BLI fans. Two independent parameters are considered to affect the design space: FPR_{BLI} and the number of fans. At first, the results are analysed for a fixed number of fans and for varying FPR and then the number of fans is also varied. The results presented herein, unless specified otherwise, refer to half of the propulsion system but the number of fans is quoted for the whole array.

For the assessment of the propulsion system with varying design considerations at mission level, different aircraft models need to be created every time. To avoid the unnecessary complexity of having to perform several iterations between the framework modules, the fuel capacity (useful LH_2) and tank weight are taken from Table 4-10. For every propulsion system sized, the estimated weight is included in the aircraft weight estimation tool and the MTOW and OWE of the aircraft are calculated. Then based on the propulsion system model and the aircraft weights the design mission is estimated with the aircraft performance module. As the fuel weight is fixed throughout the design space, the propulsion system will be assessed based on the resulting range of the design mission.

4.5.1 12-Fan assessment

Based on the ambient conditions and thrust levels as well as DP inputs discussed in Table 4-11 and Table 4-12 respectively, the multipoint sizing is performed. In Figure 4-14, the mass flow requirements of the fan closest to the centreline as well as the core are presented. In addition, the nozzle area opening of the fan at OD relative to DP is also shown. As per the example in Figure 4-8 and Figure 4-9, the higher the FPR_{BLI} , the lower the mass flow and nozzle area opening at OD. The core mass flow increases as well over increasing FPR_{BLI} due to the increasing power consumption of the BLI fans.

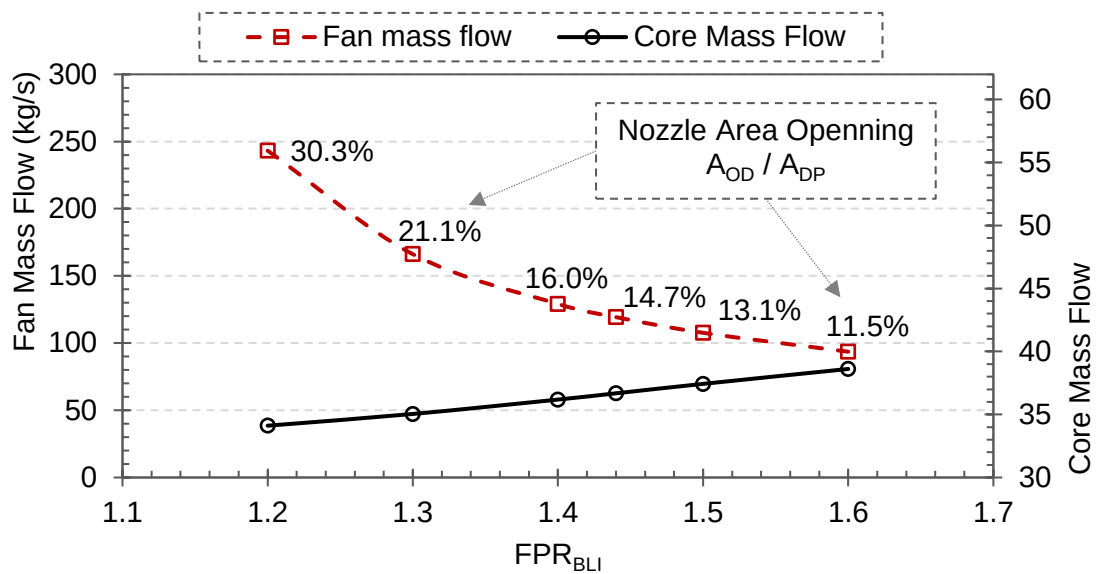


Figure 4-14 Centreline Fan and core mass flow requirements over FPR_{BLI}

Figure 4-15 shows the optimal OPR and TET selection as well as the resulting HPC exit temperature (T_3) at DP, over the FPR_{BLI} . Effectively for all the points presented, the corresponding OD TET and T_3 are matched to the restrictions included in Table 4-14. The higher core mass flow associated with the increasing FPR, the lower the allowable DP OPR and TET. For every selection of FPR_{BLI} , however, the DP TET and T_3 , although not exceeding, are relatively close to the equivalent OD limits. This is attributed to the variable nozzle area which at all times allows for optimal array fan operation. Therefore, compared with a fixed nozzle area moderated fan, this translates to power savings. The core in turn can be sized for higher TETs and OPRs that would otherwise not be possible. In the case of FPR_{BLI} of 1.2, for example, the DP TET is 1850 K throughout the take-off and climb phases of the flight and satisfies both DP and OD thrust requirements.

The propulsion system efficiency in terms of SFC is plotted in Figure 4-16, for the results obtained through the multipoint design. These are compared with another case where the TET and OPR remain unchanged, at 1800K and 85 respectively, through the different FPR_{BLI} values. Although at values in the proximity of 1.4, the set of OPR and TET selected seem to be close to the ones the multipoint design is dictating. However, at high FPR_{BLI} the SFC is

overpredicted and at low the contrary. Without accounting for multipoint correction of the TET and OPR the optimum SFC occurs at an FPR_{BLI} of 1.3, but in reality, the SFC is ever decreasing as the FPR_{BLI} decreases without showing an optimum in the examined design space.

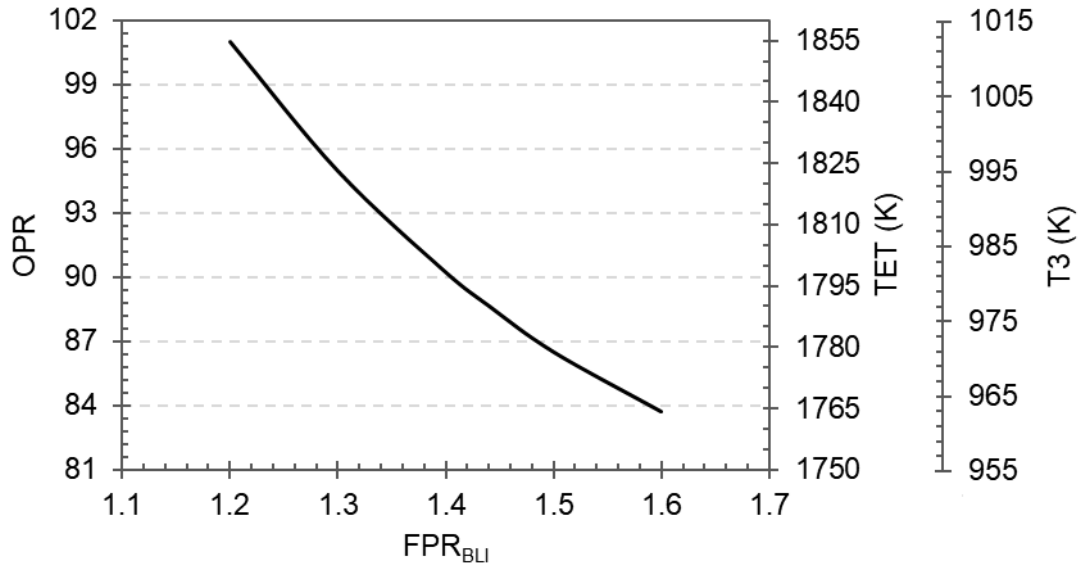


Figure 4-15 DP OPR, TET and T3 variation over FPR_{BLI} after multipoint design

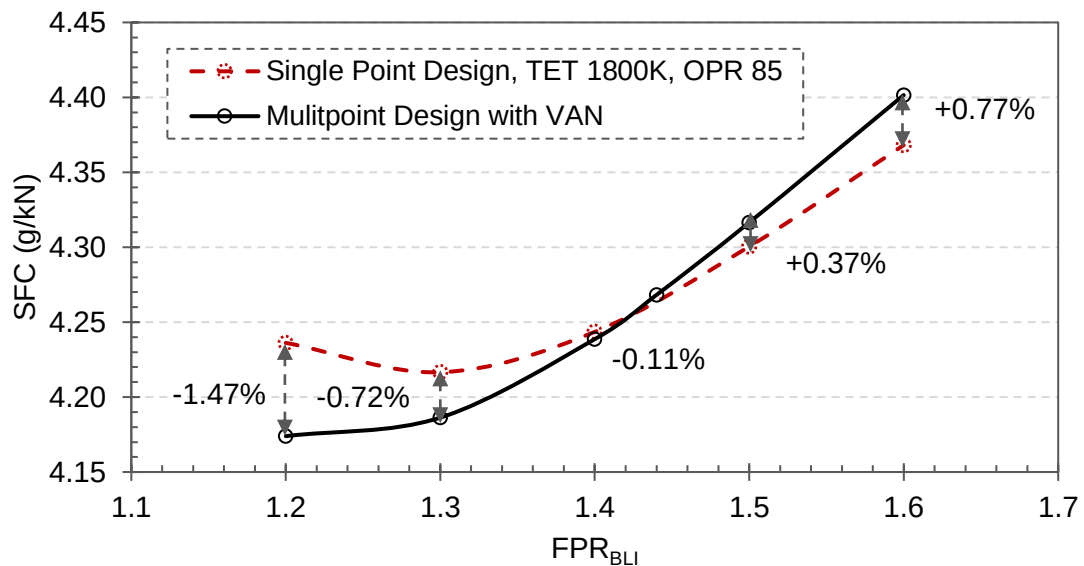


Figure 4-16 SFC against FPR_{BLI} for multipoint sizing of the cycle and for constant TET and OPR at DP

In Figure 4-17 the evolution of the specific thrust and the bypass ratio of the propulsion system is shown. Both trends depend on the overall mass flow or equivalently on the fan mass flow shown in Figure 4-14. Similarly to turbofan configurations, higher FPR_{BLI} tend to lead toward higher SFN and more compact designs. The generator power at DP and OD are presented in Figure 4-18 (top and bottom respectively) together with the annotated motor power of the centreline fan. The generator power, related to the array power requirement, appears at a low point at an FPR_{BLI} of 1.3 at DP but with no significant variance over the rest of the design space. On the contrary, at OD the power requirement not only increases significantly but also shows greater differences between the different values of FPR_{BLI} at DP. For both generators and motors the power requirement at OD are the rated requirements for their sizing.

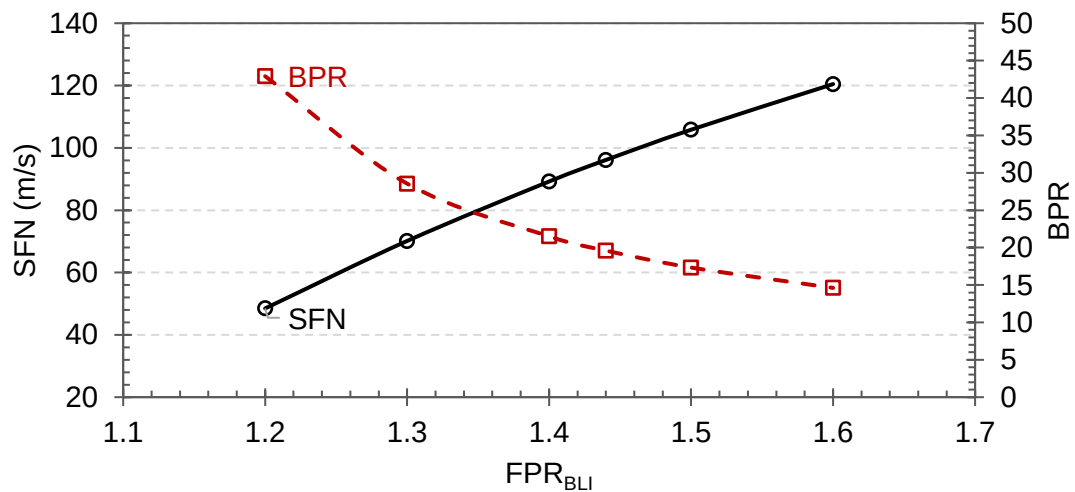


Figure 4-17 SFN and BPR over FPR_{BLI}

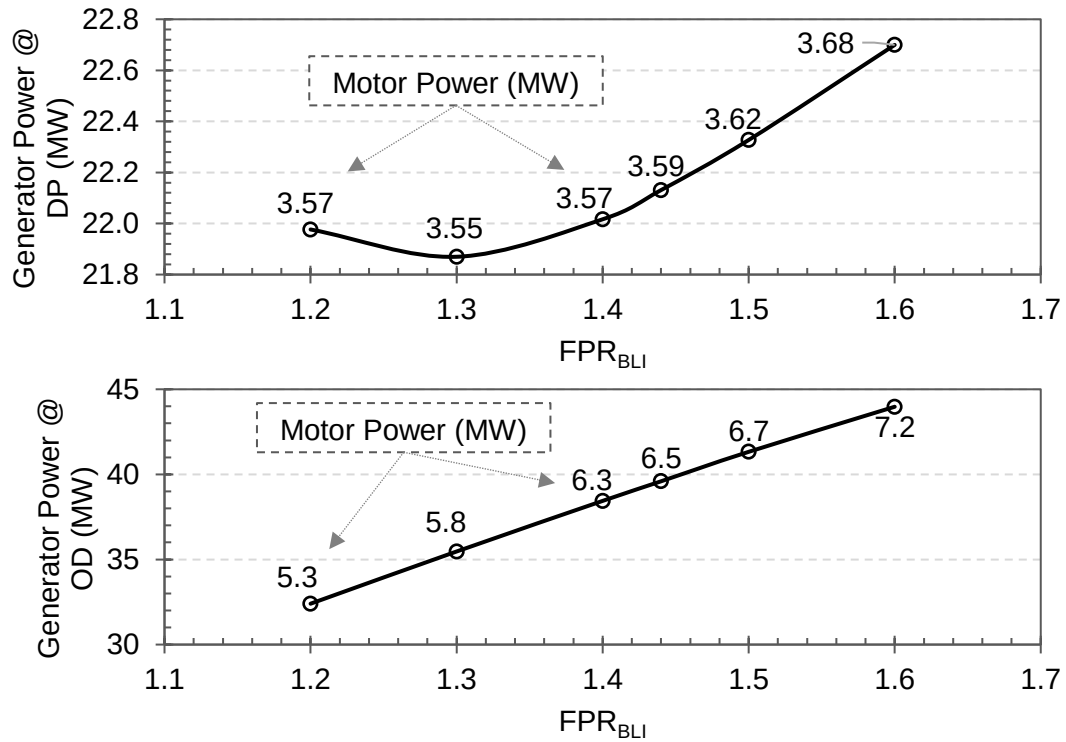


Figure 4-18 Generator power at DP (top) and OD (bottom) over FPR_{BLI}

Apart from the performance of the electrical system, the fuel preheating system can be summarised by the heat addition to the fuel against FPR_{BLI} both at DP and OD, presented in Figure 4-19. At DP the heat added to the fuel follows the general trend of the power requirement of the array. The heat dissipated from the HTS components is proportionally constant accounting for 14.2% of the overall heat added since at DP the electrical transmission efficiency does not vary significantly. At OD, contrary to the bigger variance in power requirement from the fan array, the heat added to the fuel does not show big fluctuations against the DP FPR_{BLI} and the resulting trend is due to a combination of HTS component efficiency and heat exchanger effectiveness variation.

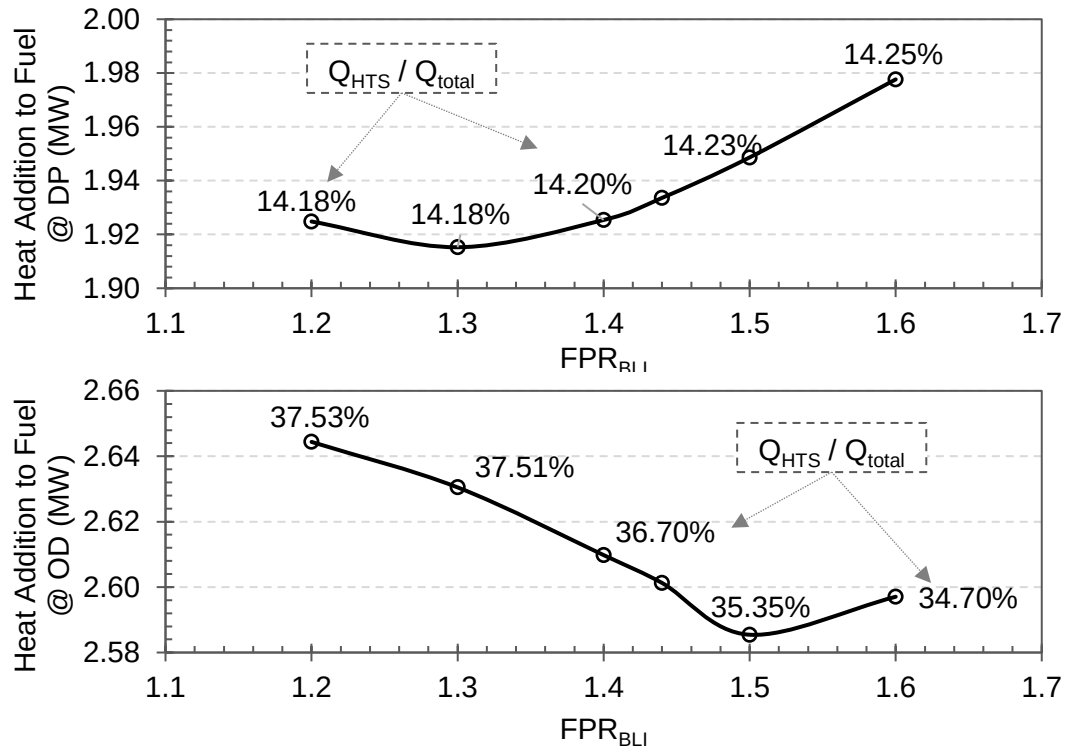


Figure 4-19 Heat added to the fuel against FPR_{BLI}

Although the heat added to the fuel is one aspect, better insight is provided by observing the fuel injection temperature as shown in plot (a) of Figure 4-20. At DP it does not only decrease by 10K from an FPR_{BLI} of 1.2 to 1.6. At OD however, it decreases drastically as the DP FPR_{BLI} is increasing. Both trends can be justified by the corresponding fuel flow evolution at DP and OD as shown in plot (b) of Figure 4-20. Although the heat added to the fuel does not vary significantly, it is the steep increase in OD fuel flow with increasing DP FPR_{BLI} that leads to a significant drop in fuel injection temperature at the same operating condition. This is also the reason why a heat exchanger effectiveness of 0.7 was selected, to allow for adequate fuel temperature throughout the flight envelope.

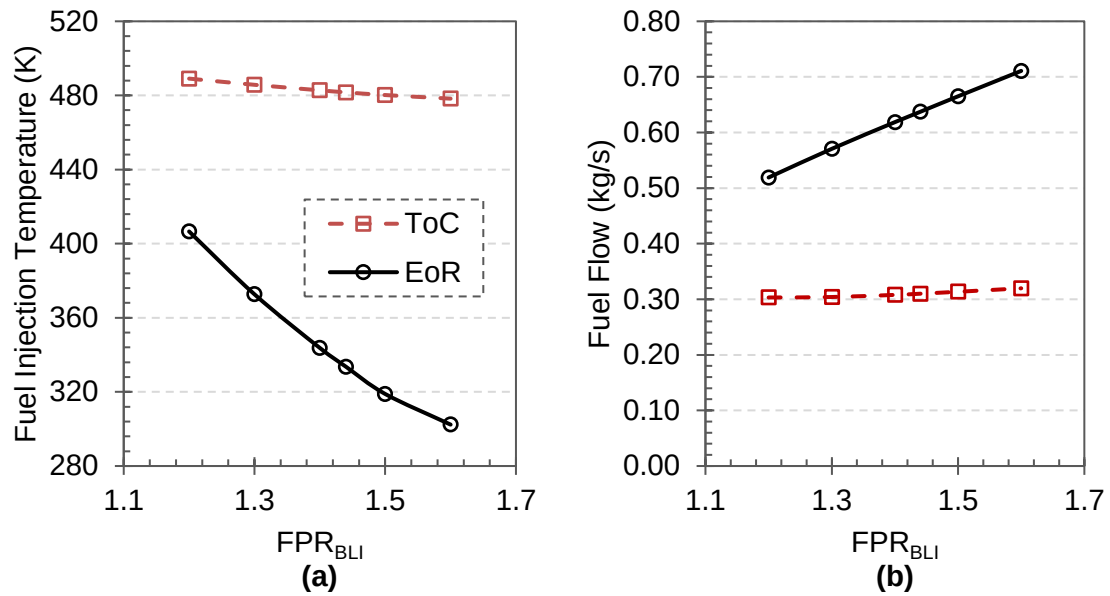


Figure 4-20 Fuel injection temperature at DP and OD against FPR_{BLI}

The plots so far are summarising to a certain extent the propulsion system performance. To discuss the weight, the shaft rotational speed selection needs to be addressed and is shown in Figure 4-21. In the graph three-shaft rotational speeds are plotted, the IP shaft, the LP – Generator shaft and each Fan – motor shaft. The IP and LP shaft rotational speeds generally decrease with increasing FPR_{BLI} , due to the increasing core mass flow requirement. On the contrary, the individual rotational speed of the BLI Fan – motor is increasing due to the decreasing mass flow requirement associated with the increasing FPR_{BLI} . Both generator and motor speeds do not show a smoothed trend due to discrete transmission ratios in the electrical system imposed by Table 4-15. Additionally, it was also observed that the speed limiting component is not consistent through the examined FPR_{BLI} . Some designs are limited by the Fans and others by the LPT and the selection was based on which design option was providing the lightest solution. It is also worth mentioning that through the developed framework that these results were obtained, the rotational speed selection only affects the weight of the system and not the performance.

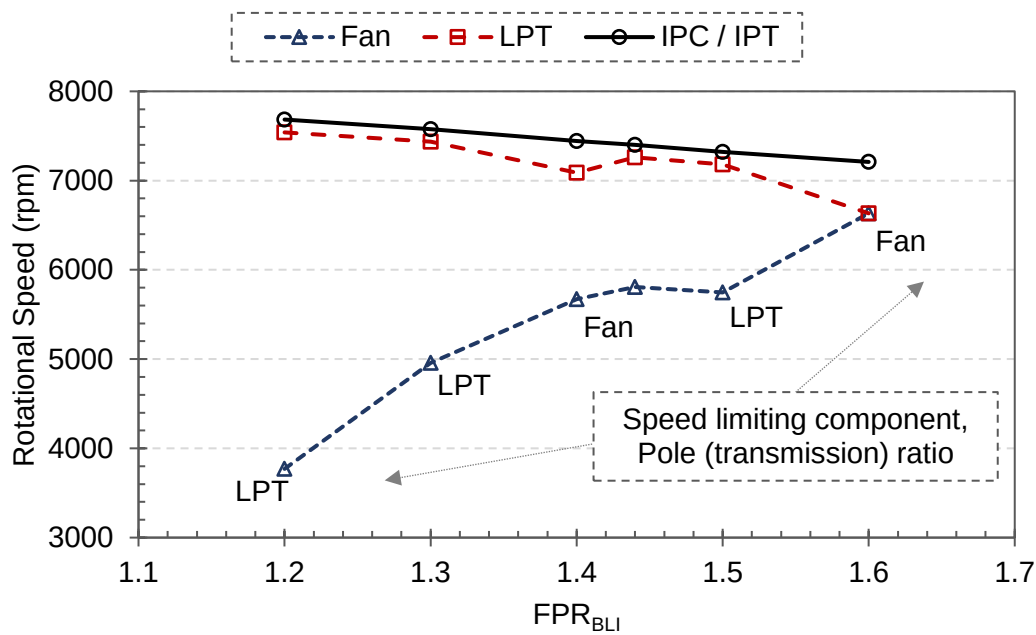


Figure 4-21 Rotational speed at DP for the IP, LP and BLI fans shafts

The weight breakdown of the major components is then presented in Figure 4-22. The fan of a single propulsor is reducing drastically with increasing FPR_{BLI} , due to the decrease in mass flow. In reverse, the weight of the turboshaft is increasing due to the higher mass flows related to the power requirement of higher FPR_{BLI} . The generator overall follows the same trend due to a slight decrease in power to weight ratio related to the combined effect of increased rated power at OD and reduced rated rotational speed at DP. The motors on the other hand are getting lighter due to the ever-increasing rated rotational speed that drives down their power to weight ratio. What can also be observed is that the kink points in the rotational speed of the LP system impose a similar behaviour to the weight of turboshaft, generator and motors.

The propulsion system weight for half the configuration (turboshaft & 6 fans), excluding the nacelles, is presented in the top plot of Figure 4-23. At the bottom plot of the same figure, the total array width for the 12 fans is shown. Accounting for the 20m limit discussed in Table 4-14, only fans with an FPR_{BLI} greater than 1.44, represented by the solid coloured line, can be accommodated. It is acknowledged, however, that this result is based on the set

of assumptions related to the fan sizing such as the fan face Mach number, hub to tip ratio and spacing between the fans.

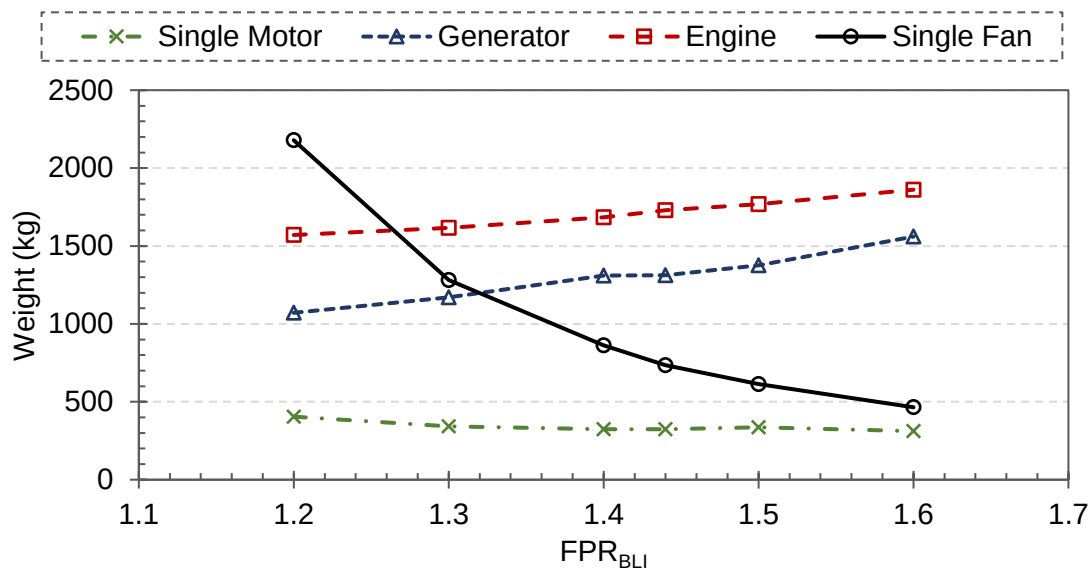


Figure 4-22 Breakdown of the component weight

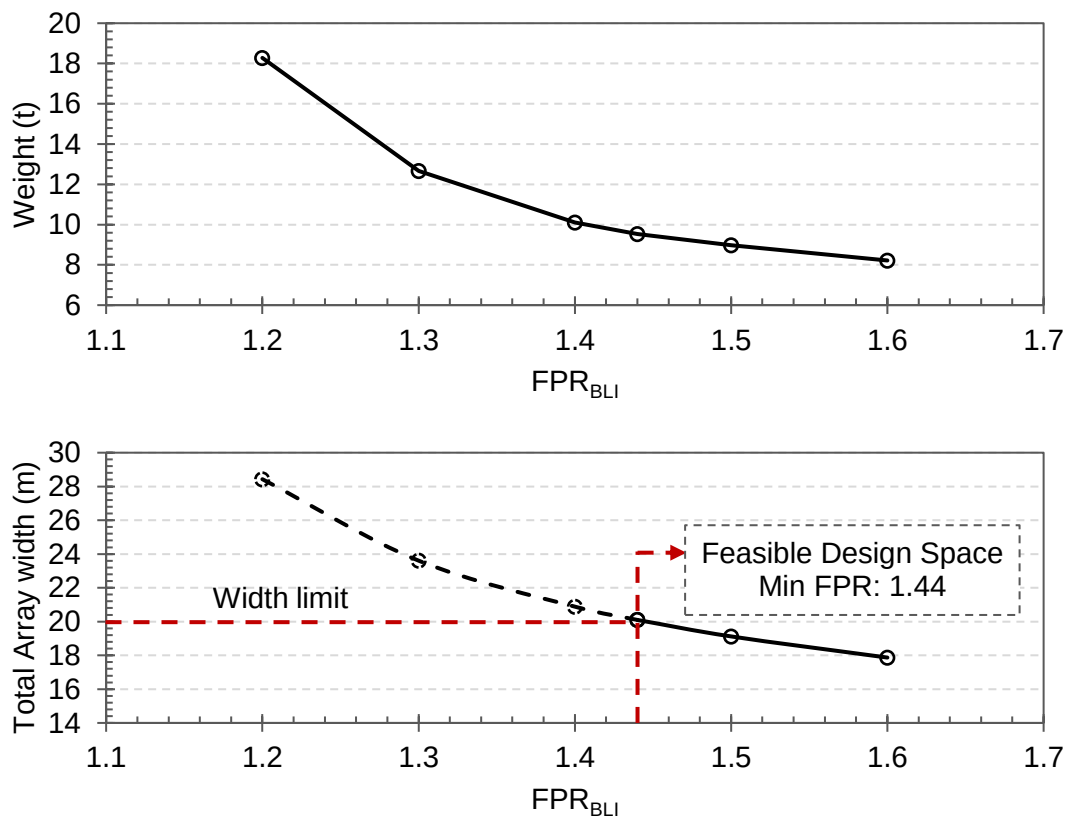


Figure 4-23 (Top) Half propulsion system weight and (bottom) total BLI array width against FPR_{BLI}

So far the multipoint sizing suggests that the most and least efficient designs (see Figure 4-16) are the heaviest and lightest respectively (see Figure 4-23), which does not provide a clear answer as to which is more efficient considering the design aircraft mission. This is the main driver of this work and can only be addressed by integrating the discussed designs with the airframe. Following the framework, a different aircraft model is created for every propulsion system configuration designed with different FPR_{BLI} . The airframe geometry and LH_2 tanks are kept the same as described in sections 4.2 and 4.3 and the difference between the aircraft designs is driven by the different propulsion system weights. Every design is then assessed based on the range of the design mission defined in section 4.2.4. The results are summarised in Figure 4-24 where the range of the different designs is plotted over OWE and MTOW of the corresponding aircraft. Notably, the maximum range can be achieved for an FPR_{BLI} of 1.4 but once the array width restriction is taken into account, 1.44 can be selected. For that value, the OWE and MTOW of the aircraft are approximately 171 and 247 respectively.

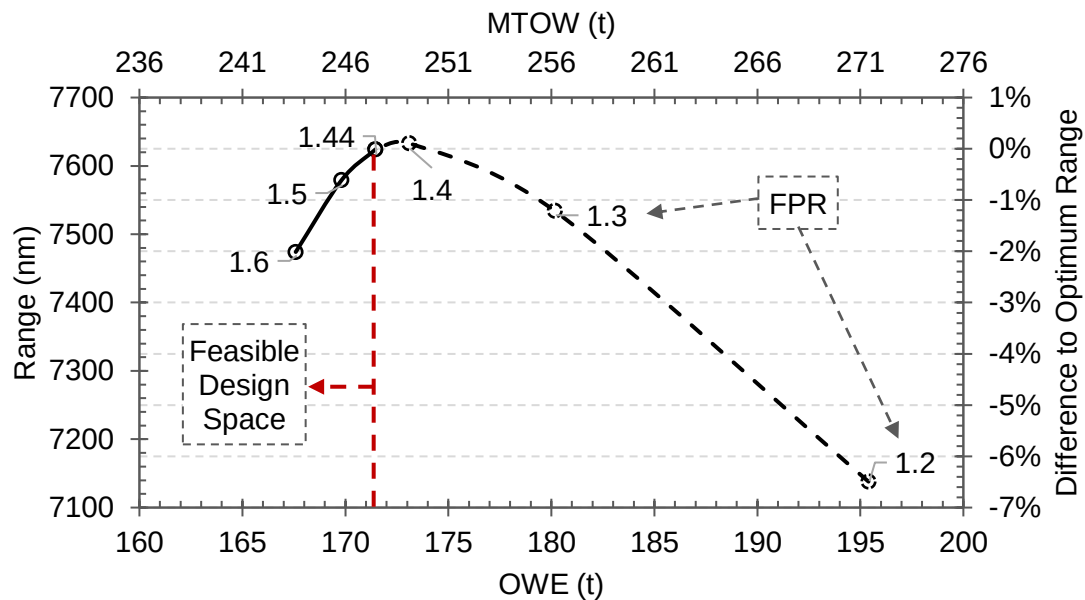


Figure 4-24 Design mission range against OWE and MTOW for the different FPR_{BLI} designs

What is also worth noting is that the FPR_{BLI} of 1.2, although delivers the lowest SFC, the added weight eventually penalises the mission range. In addition, for

FPR_{BLI} values between 1.3 and 1.5, even if the latter performs better than the former, their difference with regard to the optimum range is below 1%.

4.5.2 12-Fan with fixed nozzle area

The previous assessment showcased the results of the propulsion system sizing considering VAN for the BLI fans. An additional case is presented below to demonstrate the difference if the propulsion system were to be sized with fixed nozzle area on the BLI fans. Considering exactly the same inputs and sizing methodology the propulsion system for varying FPR_{BLI} was sized and assessed based on the design mission. Figure B-4, Figure B-5 and Figure B-6 of the appendix show DP OPR and TET, the overall propulsion system SFC and weight respectively between the fixed area nozzle and VAN (of the previous section) cases. The DP TET and OPR of the fixed area design are restricted to lower values than the VAN equivalent, especially at lower FPR_{BLI}. This in turn affects the SFC, which relative to the VAN case is by 2 – 0.2% higher for increasing FPR_{BLI}. In turn, the propulsion system becomes approximately 4.5% heavier. The combined inefficiency and added weight also penalise the range of the design mission, as shown in Figure 4-25.

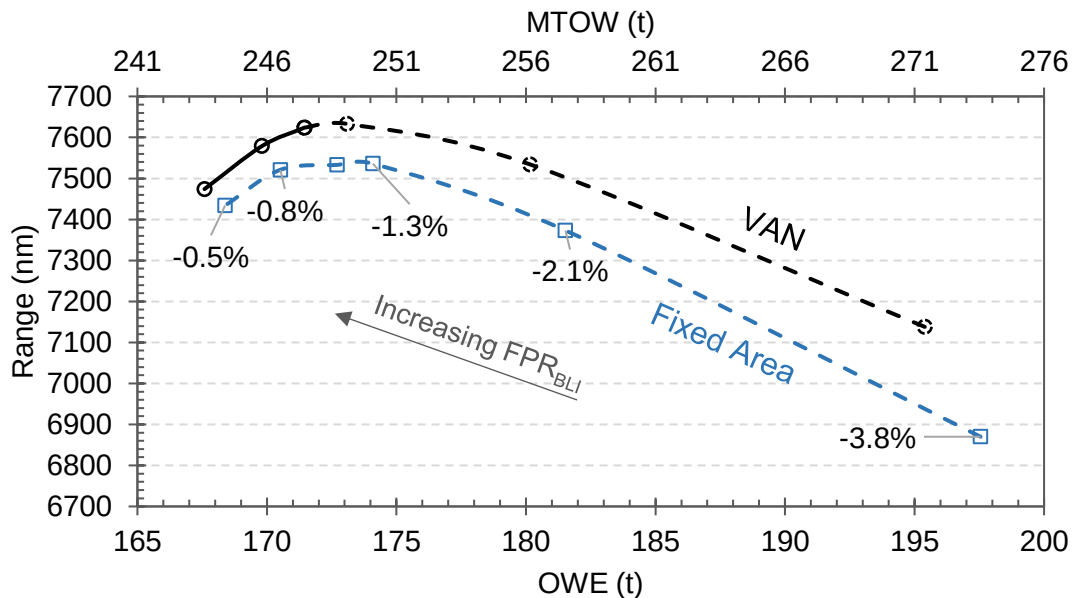


Figure 4-25 Design mission range against OWE and MTOW for the different FPR_{BLI} designs, fixed nozzle area and VNA cases

It can be observed that the lower the FPR_{BLI} the bigger benefit of the VAN and at higher FPR_{BLI} the VAN could potentially be avoided. It is also noted that the weight of the VAN is not accounted for in this context, therefore it is possible that the benefits may be marginal.

4.5.3 10 – 16 Fan assessment

So far the results for 12 fans and varying FPR_{BLI} have been presented. This section addresses the performance of different numbers of fans. As the detailed outcome and behaviour of the propulsion system parameters have been addressed in the previous section, only key results are presented. To begin with, the total fan array width is shown in Figure 4-26 over FPR_{BLI} for the different numbers of fans. As per the 20m restriction, the feasible design space is also marked. As more fans are considered, only higher FPR_{BLI} values can be used to fit in the available space and specifically for the 16 fans case no FPR_{BLI} lower than 1.7 is feasible. For the plots that will follow, the continuous line represents the feasible design space and the dashed the rest.

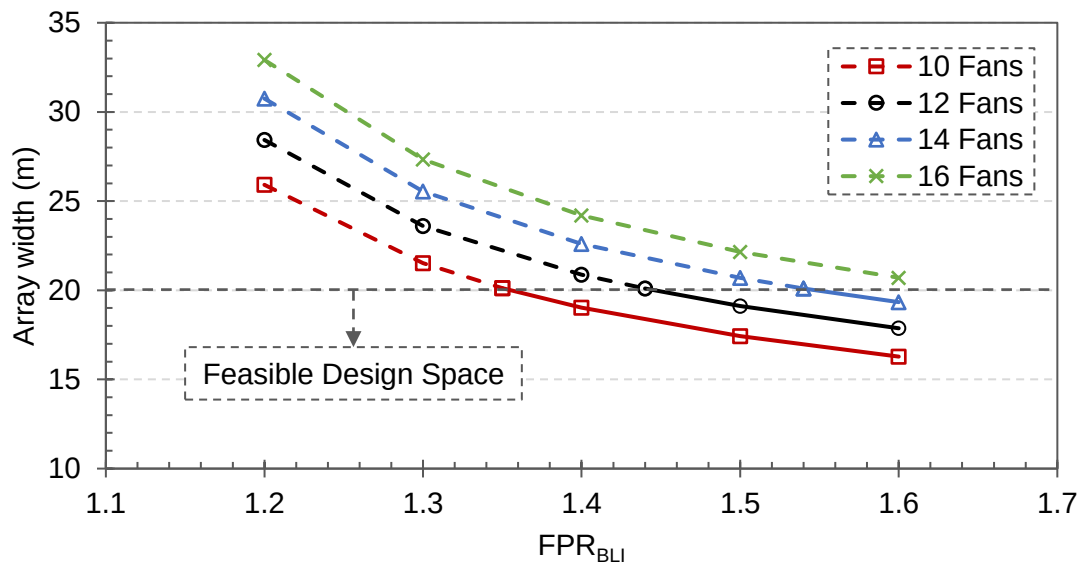


Figure 4-26 Total BLI array width over FPR_{BLI} for varying numbers of fans

The pole pair ratio between the generator and the motor is summarised in Table 4-17, for varying FPR_{BLI} and the number of fans. As shown in Figure 4-21, for

12 fans, at higher FPR_{BLI} values the Fan rotational speed is increasing and in some cases even matching that of the LPT. This can be observed in any number of fans, where for increasing FPR_{BLI} , the transmission ratio increases and in some cases suggest a speed increase between the fan and the LPT. The speed limiting component of the power transmission system is included in Table 4-18 for all the cases. Similarly to the 12 fan array discussed in the previous section, no clear pattern can be identified as to whether the LPT or the BLI Fan should be used to limit the rotational speed and dictate the transmission ratio. The power to weight ratios for the generator and motors are presented in Table 4-19 and Table 4-20 respectively. The power to weight ratio of the HTS generator shows no great variation over the design space, as the effect of decreasing rotational speed is counteracted by the increasing rated power. For the HTS motors, however, both the rated power and rotational speed increase at higher FPR_{BLI} impose a significant benefit in its power to weight ratio, so much so that the actual weight of the motor slightly decreases with increasing FPR_{BLI} as shown by Figure 4-22.

Table 4-17 Transmission ratio – generator to motor pole pair ratio

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	1/2	1/2	3/4	4/5	4/5
	12	1/2	2/3	4/5	4/5	3/3
	14	1/2	3/4	4/5	3/3	5/4
	16	2/3	3/4	4/5	3/3	3/3

Table 4-18 Speed limiting component

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	LPT	LPT	Fan	Fan	LPT
	12	LPT	LPT	Fan	LPT	Fan
	14	LPT	Fan	LPT	Fan	Fan
	16	Fan	LPT	LPT	Fan	LPT

Table 4-19 HTS Generator power to weight ratio (kW/kg)

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	30.2	30.3	28.7	29.7	29.8
	12	30.3	30.3	29.4	30.0	28.2
	14	30.3	29.5	30.2	28.1	24.6
	16	29.0	30.3	30.2	29.9	29.9

Table 4-20 HTS Motor power to weight ratio (kW/kg)

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	13.7	13.8	18.9	20.8	21.0
	12	13.1	16.9	19.4	20.0	23.0
	14	12.5	17.4	19.1	21.8	23.5
	16	14.7	17.2	18.4	22.1	22.4

The propulsion system SFC is presented in Figure 4-27. For the different numbers of fans, the overall trend appears the same over FPR_{BLI} and slightly shifts when more fans are considered. The bigger the number of fans the more efficient the system is since proportionally more boundary layer is ingested providing savings in power requirements for the same FPR_{BLI} .

The power requirement of the array only slightly changes with the number of fans, parameters such as OPR, TET, SFN, BPR_{sys} and fuel injection temperature show very little variation over the same FPR_{BLI} . These are plotted in section B.5 in the appendix. The overall weight for half the propulsion system is presented in Figure 4-28. For low FPR_{BLI} , more fans overall allow for slightly lighter propulsion systems but at higher FPR_{BLI} , other factors such as rotational speed selection (see Figure B-8 of the appendix) and allowable transmission ratios between the generator and the motors minimise the differences.

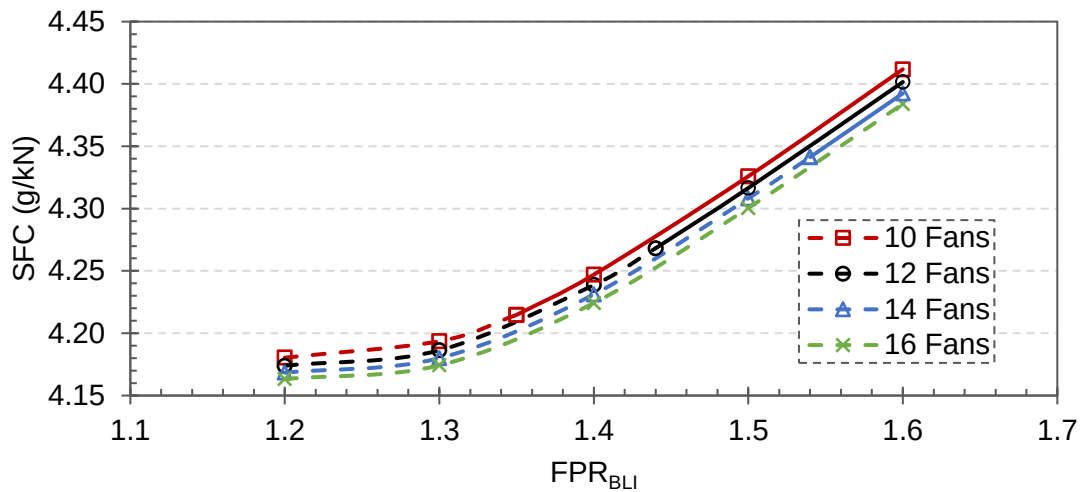


Figure 4-27 Propulsion system SFC over FPR_{BLI} for varying number of fans

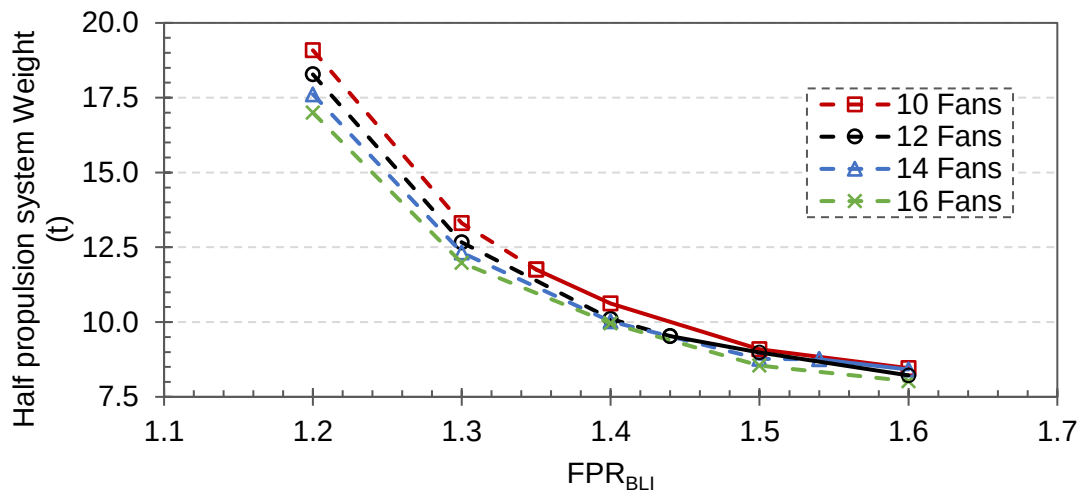


Figure 4-28 Half propulsion system weight over FPR_{BLI} for varying number of fans

Similarly, as in the 12 fan case between 10 and 16 fans the same trend evolution in SFC and overall weight over FPR_{BLI} has been demonstrated. To assess the best configuration, several aircraft were sized based on the propulsion system design space presented and their performance was assessed on the design mission range. The resulting range over OWE and MTOW is presented in Figure 4-29. Across the different numbers of fans, the ones designed for FPR_{BLI} of 1.4 achieve the maximum range, however, due to sizing restrictions, this is only feasible with 10 fans. The optimum configuration is the 12 fan array sized for an FPR_{BLI} of 1.44. In addition, most of the designs

with 1.5 FPR_{BLI} perform slightly better than the ones with 1.3. The former although approximately 3% less efficient than the latter, allow for a 10t weight saving and achieve up to 1% less range compared to the optimum feasible design, whereas the latter up to 1.5%. The designs that achieve the longest range for the different numbers of fans are summarised in Table 4-21.

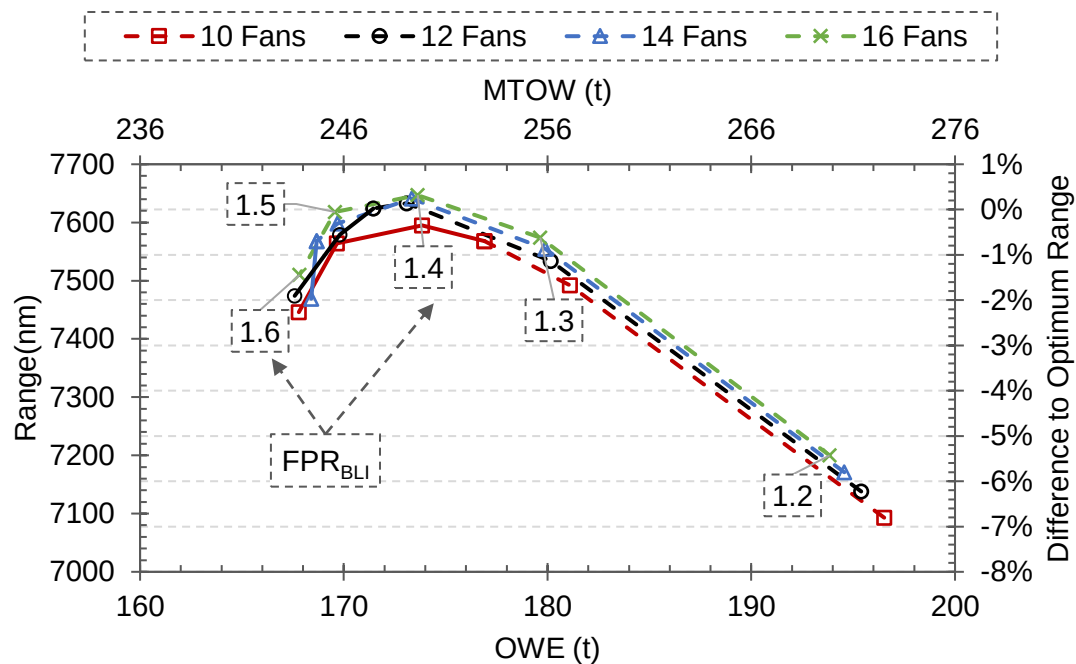


Figure 4-29 Design mission range over OWE and MTOW across varying FPR_{BLI} and number of fans

Table 4-21 Summary of optima for different number of fans

Parameter	Units	Values			
Number of Fans		10	12	14	16
FPR_{BLI}		1.4	1.44	1.54	NA
SFC (DP)	g/kN	4.25	4.27	4.34	
Propulsion system weight	Kg	10625	9530	8755	
OWE	t	173.8	171.4	168.7	
MTOW	t	249.9	247.5	244.7	
Range	nmi	7595	7624	7568	

4.6 Propulsion system design space exploration: Turbofan-powered TeDP

In the previous section, it was demonstrated how the design space of the turboshaft & BLI fans configuration is eventually restricted by the airframe integration. This section examines the replacement of the turboshaft with a turbofan. The turbofan partly produces thrust from the bypass channel and also covers the power requirement of the BLI fan array. By splitting the thrust, a two-fold effect is examined, potential fuel efficiency and weight savings as well as the coverage of the design space restricted by the maximum array width in the turboshaft case. A schematic representation of the configuration can be seen in Figure 4-30. The fan of the turbofan is connected to the LP shaft and whether a gearbox will be used will depend on the rotational speed selection and the r.

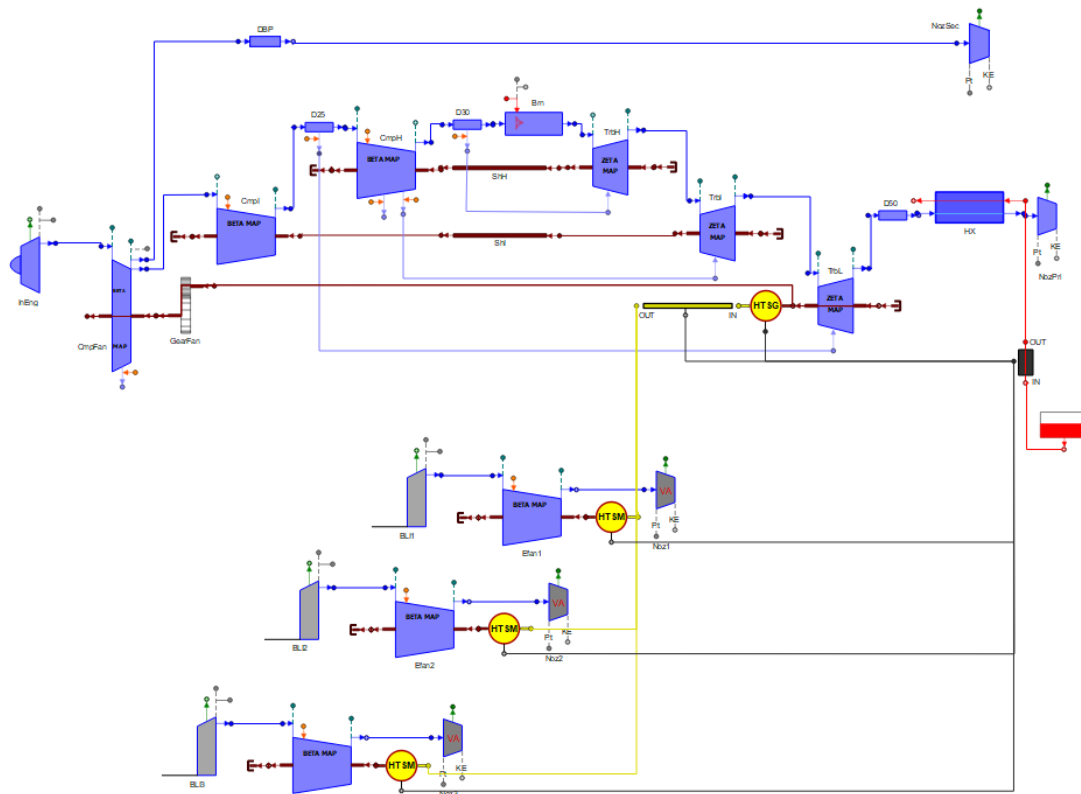


Figure 4-30 Schematic representation of Turbofan powering BLI fans in PROOSIS

In the turboshaft case it was demonstrated that apart from the sizing options that were determined by limits, two parameters were considered independent and were varied throughout the design space, the FPR_{BLI} and the number of

fans. In this configuration, the turbofan introduces an additional two, the thrust split between the bypass channel and the BLI array and the FPR_{TF} .

4.6.1 Additional inputs and assumptions

To create a reasonable comparison between the different configurations exactly the same inputs are used for this case as the ones discussed in section 4.4. With the introduction of the engine fan, additional inputs are considered:

DP related:

1. Fan isentropic efficiency is set 95% (1.5% higher compared to BLI fan efficiency as demonstrated by Gunn and Hall [149])
2. The fan core increases the pressure by 1/3 less than the bypass section
3. Bypass duct pressure losses 1.5%
4. Core net thrust 5% of the total installed thrust

Since the NPF of the BLI fans is accounting for installation effects, the contribution of the nacelle drag of the turbofan needs to be considered as well. In the context of a semi-buried installation, a semi-circular section of the nacelle is taken into account as a first-order approximation. It is sized based on the engine dimensions and its drag is calculated by eq. (3-37).

Turbomachinery gas path sizing and weight estimation

For the fan face, the same Mach number and hub to tip ratio as the BLI fans are selected as presented in Table 4-13. The Mach number at the exit of the fan core as well as the inlet of IPC and its hub to tip ratio are slightly adjusted to provide a smooth transition of the gas path between the components. The weight of the nacelle of the turbofan is calculated by the aircraft weight estimation tool in the same way as the nacelle of the BLI fan array.

4.6.2 Turbofan and BLI fan array thrust split, 16 fans case

The independent parameters of the design space can be summarised as follows:

1. FPR_{BLI}
2. Number of BLI fans
3. FPR_{TF}
4. BP and BLI array thrust split

Identifying optimum designs with four design parameters can be best performed by including an optimisation routine. However, due to the complexity of the framework and the uncertainty of the outcome a set of inputs might impose, this has been avoided. Instead, the effect of a single parameter is first examined and if the resulting SFC and weight of the propulsion system show a simultaneous increase in a certain design space, then that design space is dismissed.

In this section, the effect of the thrust split is tackled by examining the 16 BLI fan array case. In the following study, the thrust split between the BLI fans and the bypass channel is varied for different FPR_{BLI} and FPR_{TF} . The resulting SFC at DP is plotted against the thrust ratio in Figure 4-31. For increasing FPR_{BLI} , there is an increasing limit in the thrust that can be produced from the BLI fans due to the array width restriction. Similarly to FPR_{BLI} , the increasing FPR_{TF} penalises the engine efficiency and as more thrust is produced by the BP channel the heavier the influence of the FPR_{TF} becomes. For the majority of combinations between FPR_{BLI} and FPR_{TF} , the more thrust produced by the BLI array the more efficient the system. The only exception is for the case of FPR_{BLI} and FPR_{TF} of 1.6 and 1.3 respectively, where as the thrust produced by the fan array decreases, the SFC almost negligibly decreases.

For the same cases, the corresponding weight for half of the propulsion system including the nacelles is presented in Figure 4-32. The general trend is that for increasing FPR_{BLI} and FPR_{TF} the propulsion system becomes more compact and light. At the same time as more and more thrust is produced by the

Turbofan the overall weight increases. Although there are weight savings attributed to the BLI fans (smaller nacelles, motors and the generator due to their reduced power requirement), this is counterbalanced by the increase in the turbofan weight, its nacelle as well as the gearbox required to connect it to the LPT.

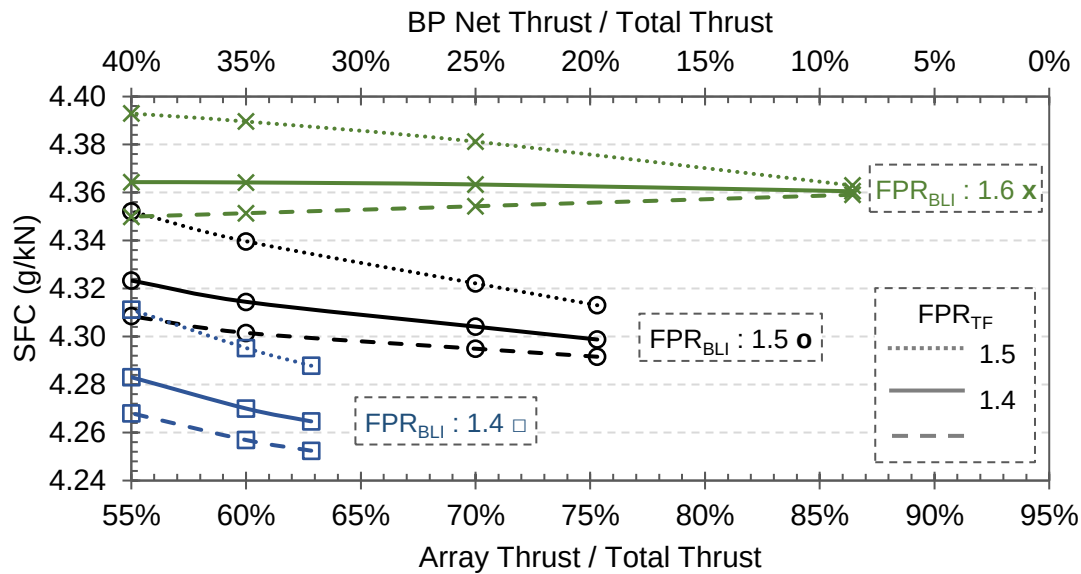


Figure 4-31 SFC over thrust split for varying FPR_{BLI} and FPR_{TF}

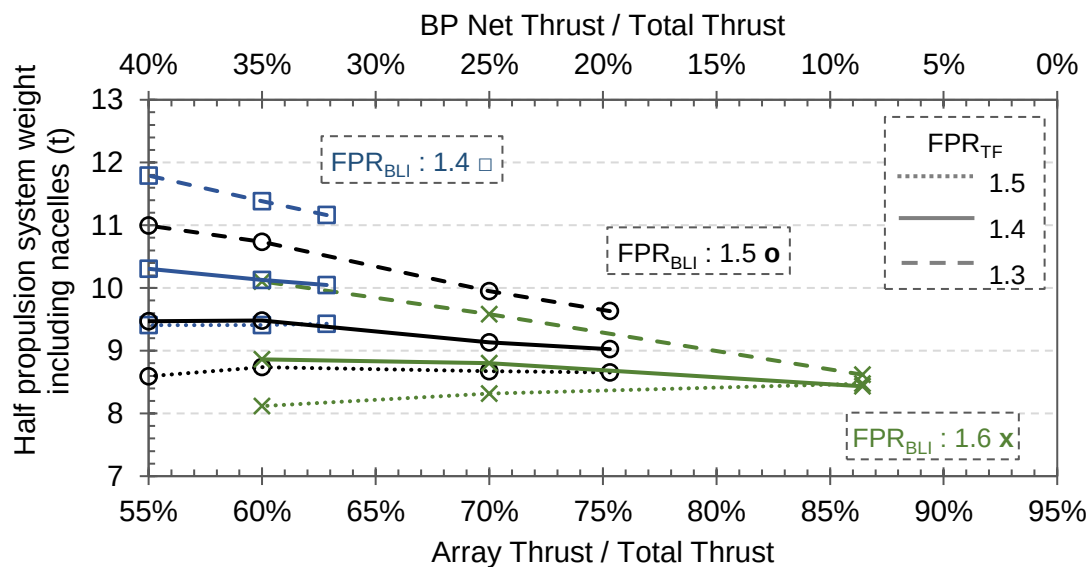


Figure 4-32 Half propulsion system weight (including nacelles) over thrust split for varying FPR_{BLI} and FPR_{TF}

Plot (a) of Figure 4-33 shows the evolution of the BPR of the turbofan (BPR_{TF}) against the thrust split and plot (b) of the same figure the BPR of the whole propulsion system (BPR_{sys}). The more thrust produced by the turbofan the higher its BPR_{TF} , while for the BPR_{sys} is driven by a combination of all the parameters involved.

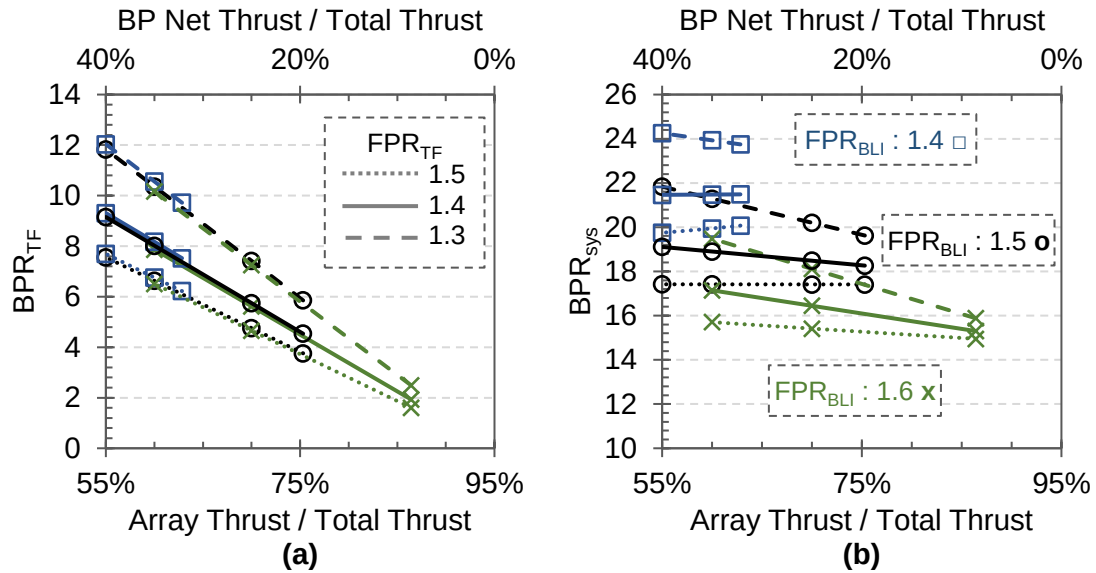


Figure 4-33 (a) BPR of the main engine and (b) overall BPR for the propulsion system over thrust split

Of all the cases examined two cases stand out, the FPR_{BLI} and FPR_{TF} of 1.6 and 1.3 and the 1.6 and 1.5 respectively. The former with more thrust produced by the turbofan shows negligible benefit in SFC but significant increase in weight, the latter negligible weight saving but significant penalty in SFC. Both cases are considered to provide maximum mission efficiency when most of the thrust produced by the BLI fans is maximised. Overall a general conclusion can be drawn that the more thrust produced by the remote fan array the more efficient and the lighter the propulsion system becomes. Therefore for the cases where the fan array cannot produce all the thrust the residual is provided by the bypass channel.

4.6.3 Turbofan FPR selection, 12 fans case

This section addresses in detail the propulsion system design space exploration for 12 fans. In the equivalent turboshaft case, in section 4.5.1, the minimum

FPR_{BLI} that can be utilised is 1.44. For every value below that, the thrust split is set to maximum so that the BLI fan array width matches the 20m restriction. For every FPR_{BLI} above 1.44, the less thrust produced by the fan array the less efficient and heavier the system is similar to the study discussed in the previous section. However, there is an exception, where when a small part of the total thrust is produced by the engine fan (in the order of 5%) the engine fan, depending on its FPR_{TF} , can become small enough, that its rotational speed matches that of the LPT. In that case, a direct drive between the two components can be achieved and a gearbox can be avoided. Effectively an additional fan is added to the configuration that does not have transmission inefficiencies and added weight compared to the electrically driven fans or a geared fan. The amount of thrust of the TF is then determined based on the minimum value that allows the Fan and the LPT to share the same rotational speed. For the 12 fans, the thrust split is presented in Figure 4-34 against the FPR_{BLI} . For every FPR_{BLI} below 1.44, an increasing amount of thrust needs to be subsidised by the Turbofan. Above FPR_{BLI} the thrust split depends on the FPR_{TF} and it increases with increasing FPR_{TF} .

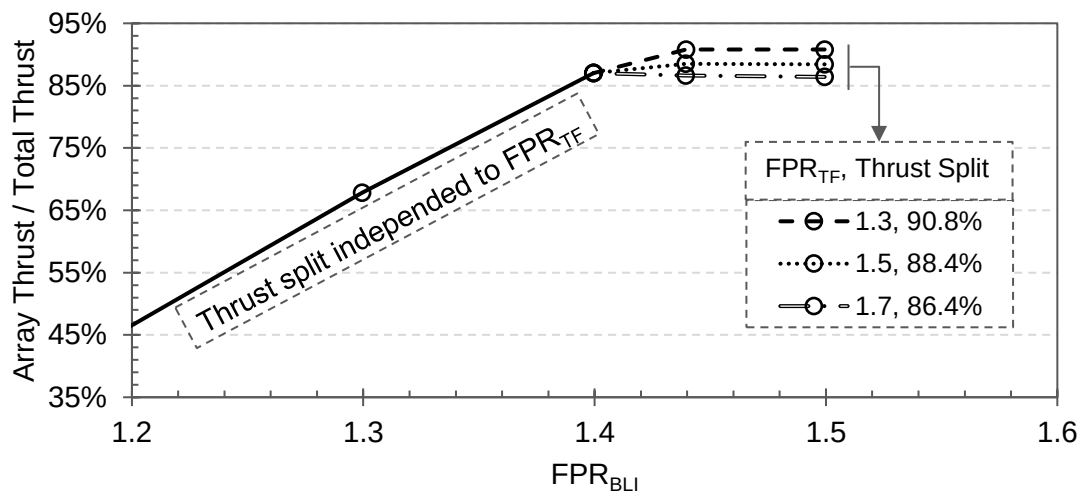


Figure 4-34 Thrust split over FPR_{BLI} for 12 BLI fans

The SFC against FPR_{TF} for different FPR_{BLI} is shown in plot (a) of Figure 4-35 and the equivalent weight for half of the propulsion system in plot (b) of the same figure. For increasing FPR_{TF} , the SFC is generally penalised, but for

FPR_{BLI} , an optimum value of 1.3 can be observed. It can also be seen that the lower FPR_{BLI} , where more thrust is provided by the turbofan, the greater the FPR_{TF} affects the engine efficiency. For the weight similarly to FPR_{BLI} the higher FPR_{TF} the more compact the turbofan and the lighter the overall system.

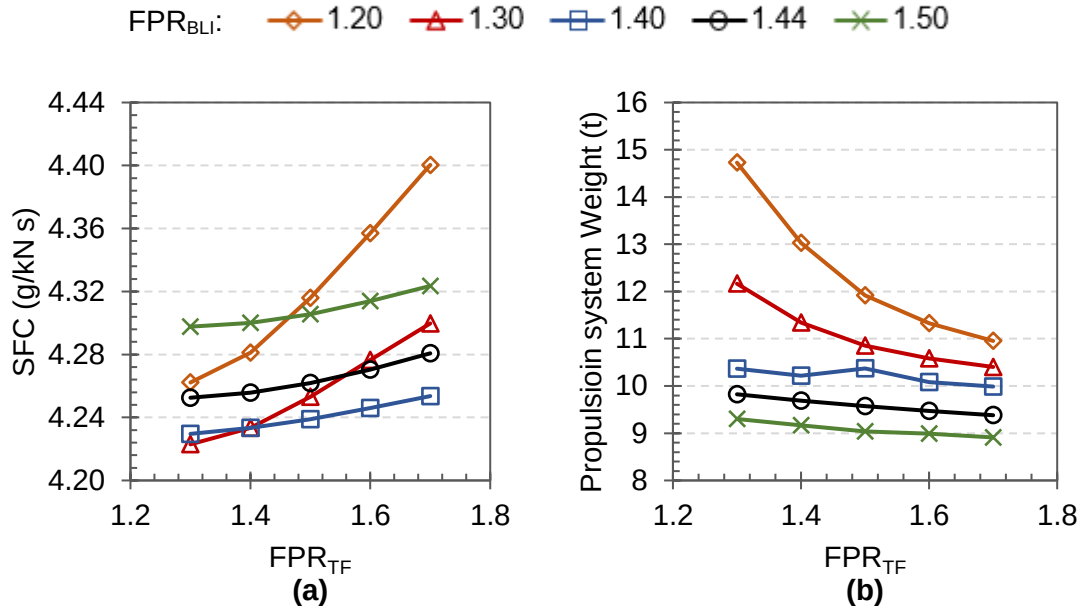


Figure 4-35 (a) SFC and (b) over FPR_{TF} for varying FPR_{BLI}

From Figure 4-35, one case that stands out is the FPR_{BLI} and FPR_{TF} of 1.4, where 87% of the thrust is produced by the BLI fan array and 8% from the BP channel. The weight of this design becomes heavier than any other choice of FPR_{TF} . A better understanding of this outcome is obtained by observing the LPT rotational speed selection presented in Figure 4-36. For the discussed set of design options, the LPT rotational speed reduces drastically compared to the other FPR_{TF} values. Figure 4-37 shows the rotational speed selection of the LPT, the fan of the turbofan and the BLI fans. The FPR_{TF} of 1.4 in combination with the thrust split imposed by the FPR_{BLI} suggest that a gearbox cannot be used based on the methodology outlined in Figure 3-37. Therefore the LPT needs to slow down to match the maximum rotational speed of the main engine fan. The rotational speed reduction, however, has a snowball effect on the whole transmission system since the generator, motors and LPT are getting heavier. The latter in particular needs to be designed with an extra stage to

avoid excessive stage loadings. For values of FPR_{TF} lower than 1.4 a geared turbofan is required whereas for higher values a direct drive.

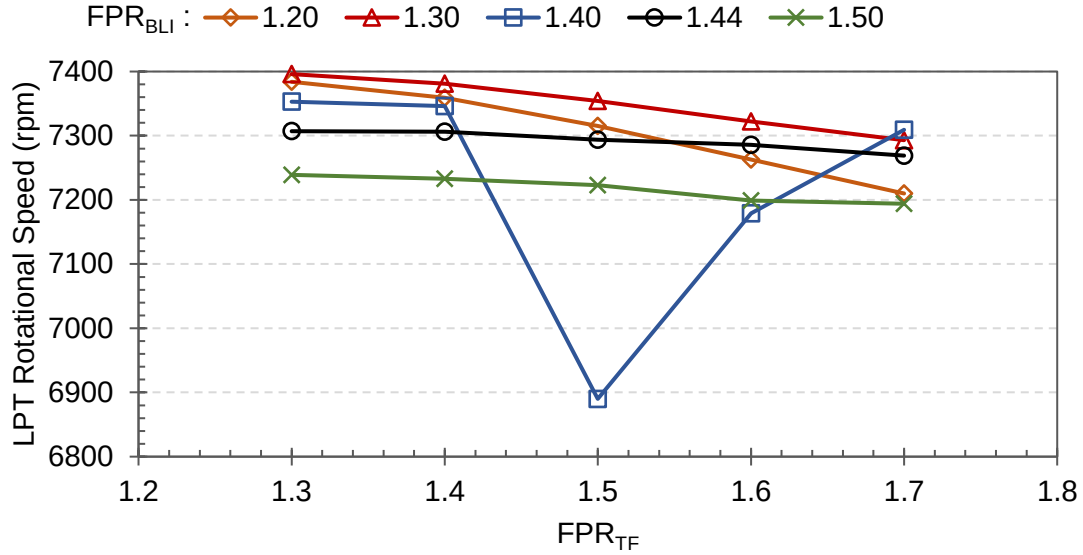


Figure 4-36 LPT rotational speed over FPR_{TF} for varying FPR_{BLI}

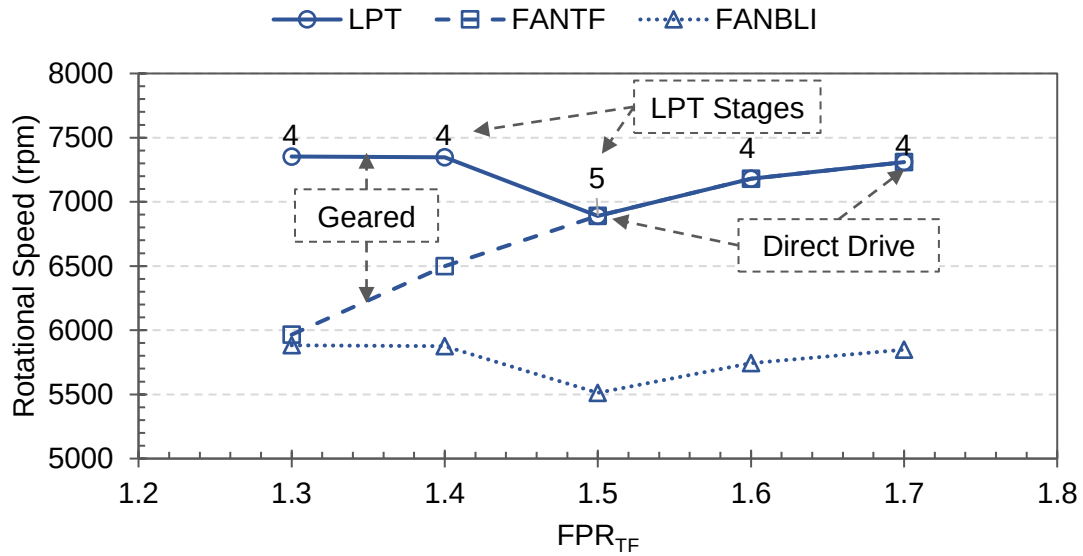


Figure 4-37 Rotational speed selection for the LPT, turbofan and array fans for FPR_{BLI} of 1.4

The gear reduction ratio for the design space discussed so far is summarised in graph (a) of Figure 4-38. For FPR_{BLI} less than 1.4, the relatively high thrust produced by the turbofan suggests that a geared configuration should be used

regardless of the FPR_{TF} of the examined range. For the rest of the design space, a direct drive turbofan is used. Lastly, in graph (b) of Figure 4-38 the weight of the electrical transmission system to the overall weight of the propulsion system is presented.

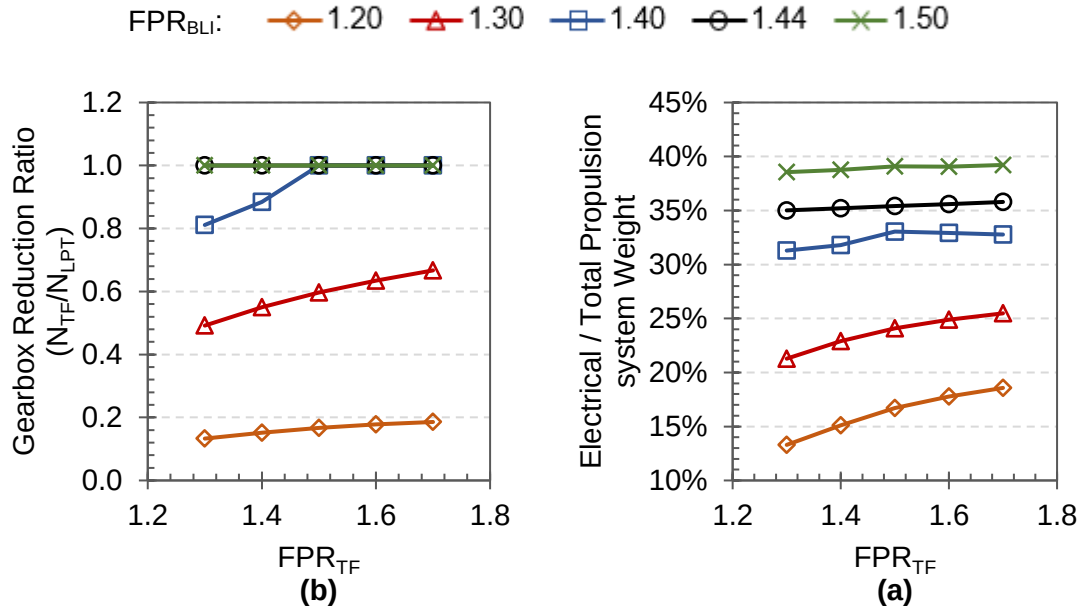


Figure 4-38 (a) Turbofan gearbox reduction ratio and (b) electrical to overall propulsion system weight ratio over FPR_{TF} for varying FPR_{BLI}

For all the designs presented different aircraft models were created and their performance on the design mission was assessed. The resulting range is plotted against FPR_{TF} for the various FPR_{BLI} in Figure 4-39. At low FPR_{BLI} values, the FPR_{TF} is showing a greater influence in the overall range due to the higher thrust provided by the TF. On the contrary, at higher FPR_{BLI} the differences in the range between FPR_{BLI} values are minimised. Overall for all FPR_{BLI} but 1.4, the FPR_{TF} of 1.5 provides the longest range.

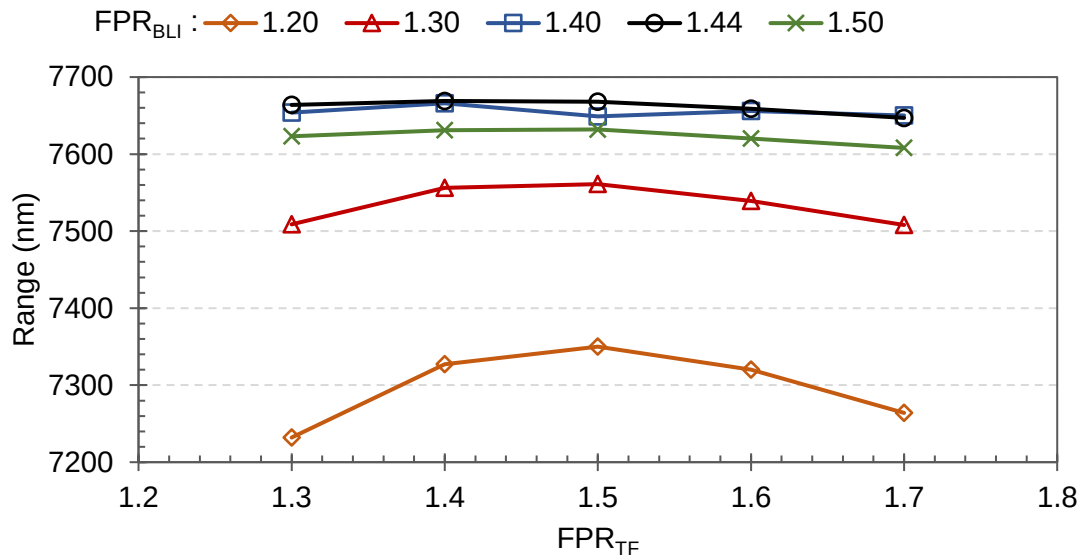


Figure 4-39 Design mission range against FPR_{TF} and FPR_{BLI}

4.6.4 Optima for 10 – 16 fan array

This section extends the design space exploration to the different numbers of BLI fans. As the thrust split and FPR_{TF} effect on the propulsion system have been addressed in the previous section, only the optimum selection of those for varying FPR_{BLI} is presented here.

The thrust ratio of the BLI fan array (FN_{BLI}) to the total thrust requirement (FN_{tot}), the turbofan gearbox reduction ratio, the SFC and propulsion system weight against FPR_{BLI} for the different number of fans are shown in plots (a), (b), (c) and (d) respectively of Figure 4-40. The more BLI fans utilised the greater the turbofan thrust requirement, mainly due to array width restriction imposed. The gearbox reduction ratio is in turn following the same trend. In addition, the more thrust produced by the turbofan the least efficient the propulsion system becomes. The latter was demonstrated in section 4.6.2 and is expanded here for different numbers of fans. For 12, 14 and 16 fans the SFC shows a minimum value for an FPR_{BLI} of 1.4. Finally, the weight of the propulsion system becomes lighter with increasing number of fans and FPR_{BLI} .

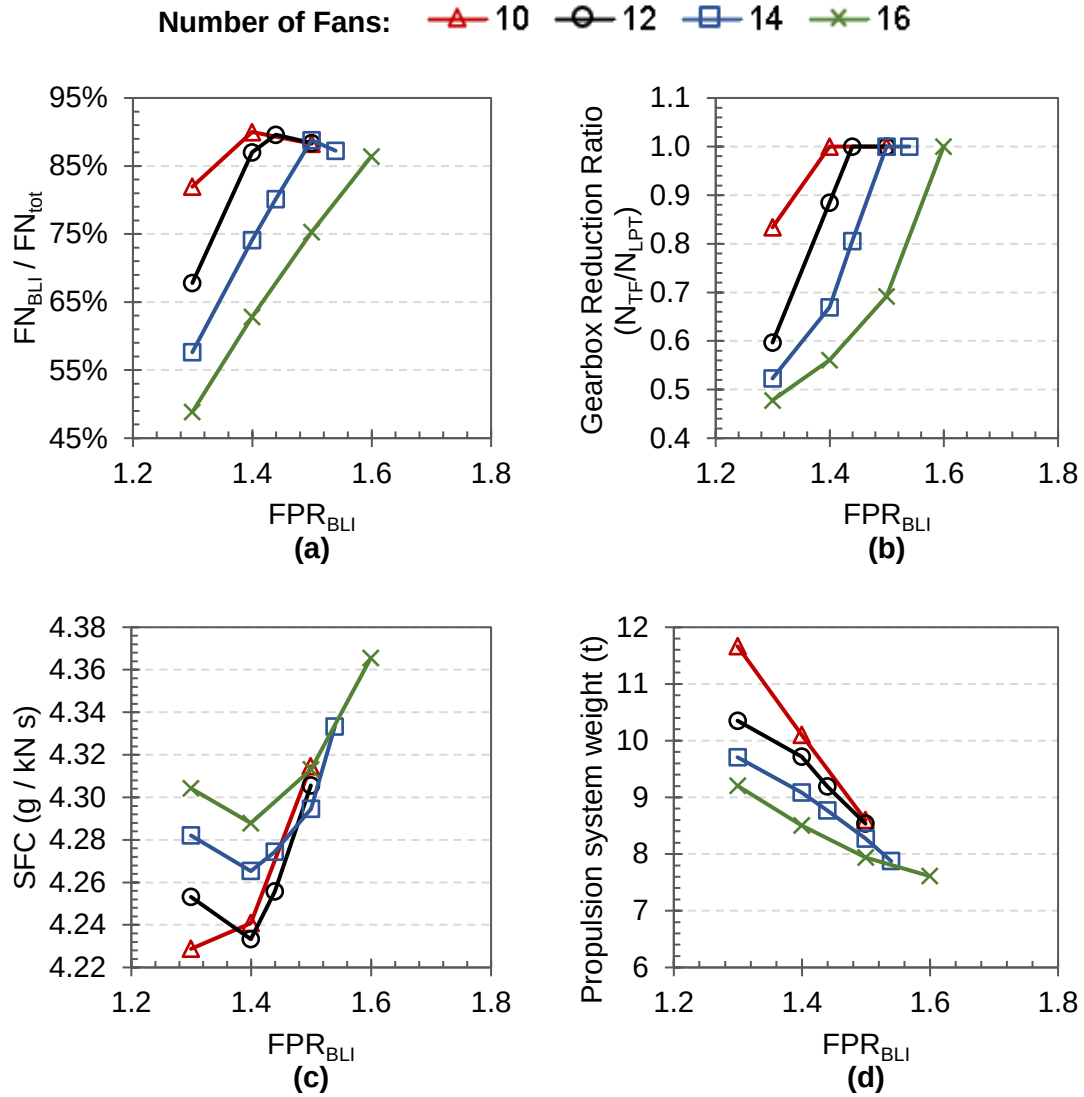


Figure 4-40 (a) Thrust split, (b) turbofan gear reduction ratio, (c) SFC and (d) Propulsion system, weight over FPR_{BLI} for different numbers of BLI fans

For the design space presented the transmission ratio between the BLI fans and the LPT, the speed limiting component of the LP system as well as the calculated power to weight ratios for the generators and the motors are presented in Table 4-22, Table 4-23, Table 4-24 and Table 4-25 respectively. The pole pair ratio maintains relatively high values that are increasing with increasing FPR_{BLI} . For the majority of the cases, sizing based on the maximum rotational speed of the LPT provided lighter solution, as it also drives the engine fan.

Table 4-22 Transmission ratio (N_{BLI} / N_{LPT}) – generator to motor pole pair ratio

		FPR_{BLI}					
		1.3	1.4	1.44	1.5	1.54	1.6
Number of Fans	10	2/3	3/4	-	4/5	-	-
	12	4/5	4/5	4/5	4/5	-	-
	14	4/5	4/5	3/3	3/3	3/3	-
	16	3/3	3/3	-	3/3	-	5/4

Table 4-23 Speed limiting component

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	LPT	BLI Fan	-	LPT	-
	12	LPT	LPT	LPT	LPT	-
	14	LPT	LPT	BLI Fan	BLI Fan	LPT
	16	LPT	LPT	-	LPT	-

Table 4-24 HTS Generator power to weight ratio (kW/kg)

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	29.5	29.2	-	29.8	-
	12	28.6	30.0	30.1	29.9	-
	14	27.7	29.1	28.1	28.7	29.7
	16	26.8	28.3	-	29.1	-

Table 4-25 HTS Motor power to weight ratio (kW/kg)

		FPR_{BLI}				
		1.2	1.3	1.4	1.5	1.6
Number of Fans	10	17.1	19.1	-	20.7	-
	12	17.9	19.5	19.8	19.7	-
	14	16.2	17.8	21.1	22.0	22.7
	16	17.5	19.4	-	20.8	-

For all the design options discussed above, the corresponding design mission range is then presented in Figure 4-41, against the OWE and MTOW of every aircraft. Compared to the optimum design of the turboshaft configuration, the majority of the design options achieve a greater range, but only with a small difference of up to +0.6%. In more detail, every optimum configuration for the

different number of fans is summarised in Table 4-26. For the majority of the design space, imposing the maximum speed of the LPT provided for the lightest solution especially because it was coupled with the engine fan. There were, however, some designs where the combination of thrust split and FPR_{BLI} would allow the BLI fan maximum speed to be the limiting parameter. This was the case for the propulsion system that provided the maximum range for 10 and 14 fans. Comparing the design mission range between the turboshaft and turbofan configurations for the same number of fans, an average improvement of 0.9% can be observed. Out of all the designs presented, the one with 12 fans, FPR_{BLI} of 1.44 and FPR_{TF} of 1.4 provides the longest range of 7669 nmi and will be chosen to assess the performance of the hydrogen aircraft.

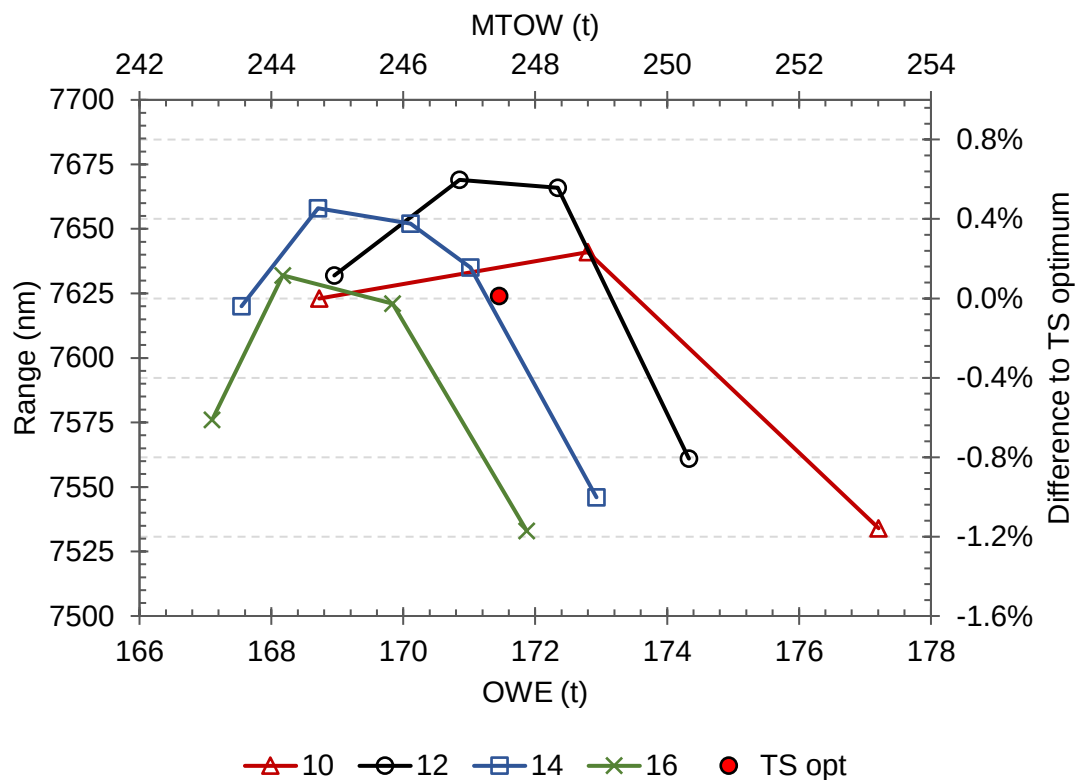


Figure 4-41 Range over OWE and MTOW for varying number of fans

Table 4-26 Summary of optima for different numbers of fans

Parameter	Units	Values			
Number of Fans		10	12	14	16
FPR_{BLI}	-	1.4	1.44	1.5	1.5
FPR_{TF}	-	1.5	1.4	1.5	1.5
FN_{BLI} / FN_{tot}	-	90%	89.6%	88.8%	75.3%
SFC (DP)	g/kN	4.24	4.26	4.29	4.31
Speed limiting Component	-	BLI Fan	LPT	BLI Fan	LPT
Turbofan configuration	-	DD	DD	DD	Geared
Propulsion system weight	kg	10100	9189	8272	7939
OWE	t	172.8	170.9	168.7	168.1
MTOW	t	248.9	246.9	244.8	244.2
Range	nmi	7641	7669	7658	7632
Improvement in Range compared to equivalent TS configuration with same number of BLI fans	-	0.96%	0.59%	1.19%	NA

4.7 Final aircraft and mission performance

As the propulsion system has been optimised this section discusses the overall aircraft characteristics and performance. An illustration of the sized aircraft is presented in Figure 1-1.

4.7.1 Aircraft weight breakdown

The aircraft breakdown of the different weight groups in terms of their actual weight as well as fractions of the MTOW are presented in Table 4-27. As hydrogen is much lighter than Jet A-1 it only accounts for 11.5% of the MTOW. The rest of the component fraction weights become more significant than kerosene powered aircraft and this is adequately summarised by the OWE to MTOW ratio. For this design is 69.2%, which is in within the 60-70% range predicted by previous conceptual studies of LH₂ aircraft [15, 16, 20, 22, 25, 26]. Although this value depends on the design payload and range, it is significantly more than the 45-55% a typical long-range kerosene aircraft would have.

In extension, the propulsion and fuel system (including LH₂ tanks) becomes one of the major contributors to the aircraft weight at 14.2% of the MTOW (9% for propulsion system and 5.2% for fuel system). In comparison, for short to medium range and long-range kerosene fuelled aircraft, the equivalent fraction is 8.3% [108] and 6.2% respectively (estimated according to Table 3-2). The difference is mainly attributed to the fuel tanks, where contrary to the LH₂, the Jet A-1 tanks are synergistically optimised to be part of the structure of the wing. Overall this highlights the importance of prioritising the lightweight designs in the sizing process. A more detailed breakdown of the component weight of the aircraft can be found in Table B-3 of the appendix.

Table 4-27 Aircraft weight breakdown

Weight Build-up	Constituent kg	Part Fracs
Structures	78659	31.9%
Integrated Propulsion & Fuel System	34954	14.2%
Landing Gear	12500	5.1%
Non-Propulsive	12703	5.1%

Weight Build-up	Constituent kg	Part Fracs
Furnishings, Equipment & Operating Items	32036	13.0%
OWE	170852	69.2%
Payload	47610	19.3%
Zero-Fuel Weight	218462	88.5%
Release Fuel (less Ramp Fuel)	28356	11.5%
MTOW	246818	100.0%

The characteristic maximum weights of the aircraft are summarised in Table 4-28. The maximum landing and zero fuel weight, determined based on the maximum payload mission, are slightly less than the MTOW. This in fact can be a limiting parameter on the maximum payload selected for hydrogen aircraft and also drives the weight of the undercarriage of the aircraft.

Table 4-28 Aircraft characteristic weights

Parameters	Units	Values
Maximum Ramp Weight (MRW)	(t)	246.9
Maximum Take-off Weight (MTOW)	(t)	246.8
Maximum Landing Weight (MLW)	(t)	240.9
Maximum Zero Fuel Weight (MZFW)	(t)	238.3

4.7.2 Propulsion system

As the key features of the propulsion system have been covered by the design space exploration section, here a detailed insight is provided for four representative conditions across the design mission, the ToC, EoR, design mid-cruise and end of descent. The performance is summarised in Table 4-29 and is divided into several sections: the ambient conditions, turbofan, average BLI fan, electrical transmission and fuel preheating and overall propulsion system performance.

For the turbofan, the low thrust output in combination with the FPR_{TF} , allows for a direct coupling of the engine fan with the LPT and a relatively low engine BPR of 1.27 at ToC. The high OPR values allowed by the T3 limit at EoR in combination with VAN, dictate a relatively high T3 at cruise as well, which from an engine lifing point of view might not be desirable. The TET on the other hand, although is maintained relatively high during take-off and climb, shows a bigger drop during cruise compared with the T3. The turbofan thrust at ToC accounts for 10% of the total thrust requirement and is divided almost 50-50 by the core and the BP gas paths. Since different operating maps are used to predict the core and BP fan performance both the BPR_{TF} as well as the thrust split differ over the operating conditions.

For the BLI fans, the inlet average pressure and velocity loss at ToC are 4% and 4.8% respectively and range between $\pm 1\%$ and $\pm 1.5\%$ depending on the installed location. At cruise where the aircraft speed is slightly higher and the mass flow requirement is lower the pressure and velocity losses are slightly bigger. Every fan approximately accounts for 15% of the total thrust (accounted for half the propulsion system). The nozzle area, since it is designed at ToC is not required to greatly vary for cruise. At EoR, however, it needs to allow for 13% area increase.

The electrical transmission system provides a 0.8 transmission ratio between LPT and the BLI fans which dictates 4/5 pole pair ratio between the generator and the motors. The transmission efficiency is relatively high throughout the operating conditions due to the HTS technology adopted and the hydrogen cooling. If LH_2 is not used as fuel then both the weight and the power requirement of the cryocoolers need to be accounted for as in [28, 67]. The heat disposed from the HTS machines accounts approximately for 14% of the total heat added to the fuel at ToC and increases at EoR. The fuel injection temperature mainly driven by the heat exchanger performance and the core power setting shows significant variation between EoR and end of descent in excess of 200K. An interesting number that puts into perspective the fuel

preheating requirements, is that the overall heat addition to the fuel is 5% compared to the total heat released in the combustor.

Overall the propulsion system has an installed SFC of 4.16 g/kN at mid-cruise which is equivalent to 500 kW/kN of thermal energy. The BPR_{tot} at the same operating point is 20.2 and 90% of the thrust is almost constantly produced by the BLI fan array. The SLS conditions are not included as the EoR was predominantly used for the sizing, however, it is worth mentioning that **the rated thrust at ISA conditions is 384kN.**

Table 4-29 Propulsion system performance of representative operating conditions

Parameter	Units	Values			
Operating condition		ToC	EoR	Mid Cruise	Descent
Ambient					
Mach Number	-	0.83	0.25	0.83	0.35
Altitude	ft	40000	0	40000	3000
ISA Deviation	K	10	15	0	0
Engine					
Mass Flow	kg/s	66.47	167.92	65.42	82.69
Fuel Flow	kg/s	0.2466	0.6402	0.2142	0.1006
BPR_{TF}	-	1.27	1.31	1.31	2.11
Combustor Exit Fuel/Air ratio	(x1000)	9.24	9.72	8.29	5.13
OPR	-	92.1	64.6	85.7	15.9
T3	K	979	1030	912	679
TET	K	1791	1850	1660	1200
Generator power output	MW	16.7	37.1	14.6	4.1
Core Nozzle Pressure ratio	-	1.42	1.11	1.37	1.01
Core Nozzle Area	m ²	0.778			
BP Nozzle Pressure ratio	-	2.15	1.30	2.10	1.12
BP Nozzle Area	m ²	0.392			
BP net thrust	kN	2.9	12.5	2.6	0.7
BP to Total thrust ratio	-	5.0%	5.2%	5.0%	3.2%
Core Net Thrust	kN	2.90	9.38	2.01	-1.95
Core to Total thrust ratio	-	5.0%	3.9%	3.9%	-8.4%
HP Spool Speed	rpm	15604	15441	14683	11837
IP Spool Speed	rpm	8158	7563	7662	3655

Parameter	Units	Values			
Operating condition		ToC	EoR	Mid Cruise	Descent
LPT – Fan Speed	rpm	7276	6654	6901	3187
Fan					
Mean Mass Flow	kg/s	89.1	263.1	88.9	124.0
Fan Pressure ratio	-	1.45	1.27	1.41	1.05
Mean Power Requirement	MW	2.73	6.15	2.38	0.64
Mean Installed Thrust	kN	8.6	36.3	7.8	4.0
Mean BLI Fan to Total Thrust Ratio	-	15%	15%	15%	17%
Rotational Speed	rpm	5821	5323	5521	2550
Mean Intake Pressure Losses	-	4.3%	0.7%	4.3%	1.4%
Mean Velocity loss	-	5.0%	2.4%	5.1%	6.3%
Nozzle exit area	m ²	0.960			
Nozzle area opening relative to design	-	0%	11%	-1%	-13%
Transmission / Fuel heating system					
Overall Transmission Efficiency	-	98.7%	98.5%	98.7%	98.5%
LPT to BLI Fan	-	80.0%	80.0%	80.0%	80.0%
Fuel Injection Temperature	K	513	340	490	578
Total Heat Input to Fuel	MW	1.65	2.66	1.36	0.78
Electrical transmission Heat Contribution	-	13%	21%	14%	8%
Total Heat Input to Fuel to Combustor Heat release	-	5.6%	3.5%	5.3%	6.4%
Overall					
Net Thrust	kN	57.8	240.0	51.5	23.2
SFC	g / kN s	4.26	2.67	4.16	4.33
Specific Energy Consumption	kW/kN	512	320	499	520
Specific Thrust	m/s	96.1	137.4	85.8	28.0
System Bypass ratio	-	19.6	23.0	20.2	30.2
array thrust split	-	90%	91%	91%	104%
Propulsive efficiency	-	85%	56%	85%	90%

Finally, the weight breakdown of the components of the propulsion system is presented in Table 4-30 and the cross-sectional view of the coupled turbofan – generator and BLI fan – motor in Figure 4-42. From the schematic it can be observed that due to the low bypass ratio of the engine and the size of the LPT, the bypass flow path will require a complex geometry exhaust duct. In addition,

the size of both the HTS generator and motor compared with the turbofan and the BLI fan respectively can be appreciated. Although it is possible to size the generator with a small enough diameter to fit below the core exhaust gas path, a long exhaust duct would be required to accommodate the installation. Alternatively, the generator may be installed perpendicular to the direction of the flow and buried in the fuselage. This would allow for bigger diameters that reduce the weight of the generator.

Table 4-30 Propulsion system weight breakdown

Component	Units	Values
Turbofan	kg	1863
Generator	kg	1244
Fan (each)	kg	682
Motor (each)	kg	309
Total	kg	9189

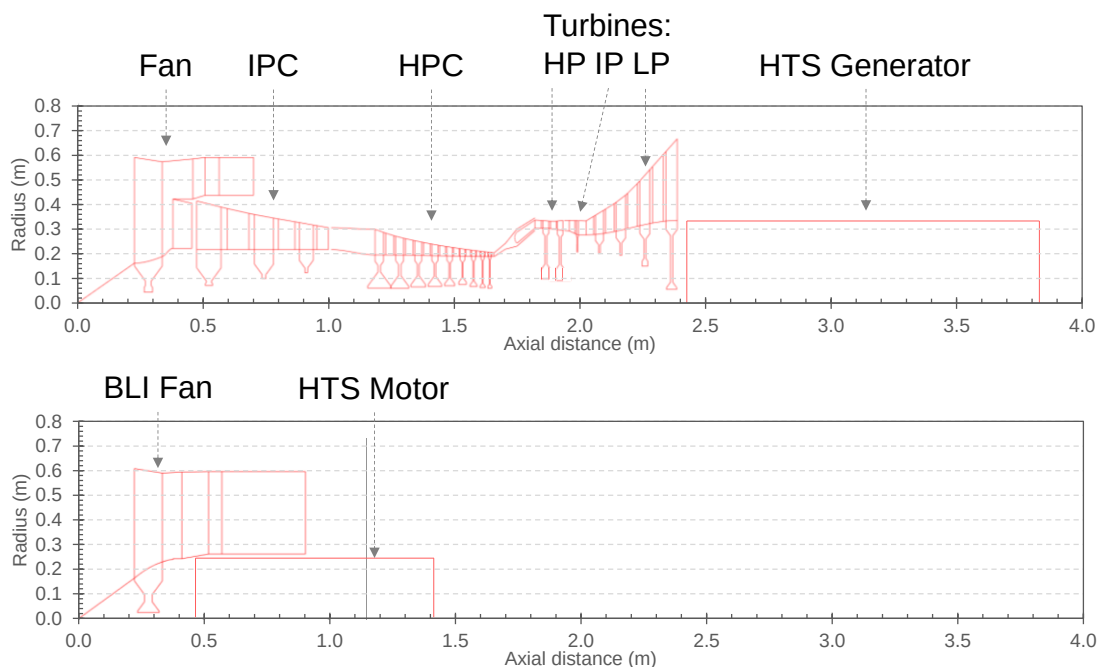


Figure 4-42 Cross sectional view of the turbofan coupled with the generator (top) and BLI fan with motor (bottom)

4.7.3 Aircraft performance

The final section summarises the performance of the aircraft with a few representative off-design missions. Figure 4-43 presents the payload to range capabilities of the aircraft based on the dT_{ISA} profile considered in the design mission as well as at ISA conditions. As in both cases, the cruise segment was at ISA conditions only minor differences between the two profiles occur. Compared to the maximum fuel mission (the design mission), the maximum payload mission shows a significant reduction in range, greater than the one a typical kerosene aircraft would have. This comes in agreement with previous findings [25] and is attributed to the fact that at MTOW, as less LH_2 mass is carried on board the available energy is reducing drastically. If a greater range at maximum payload is required, then the whole aircraft needs to be redesigned for a higher design payload.

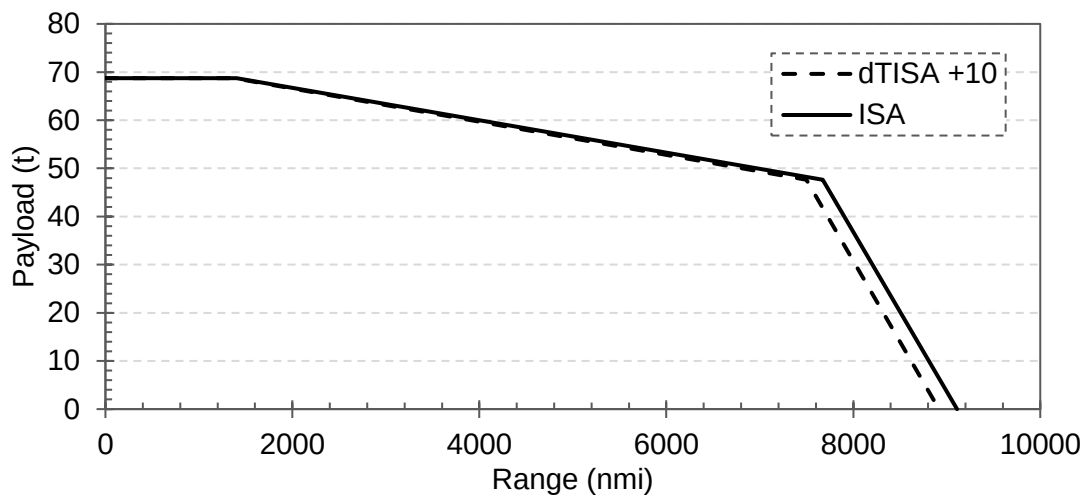


Figure 4-43 Payload – range diagram

Figure 4-44 shows the off-design performance of the aircraft for the typical payload in terms of fuel burn and energy per passenger nmi against the mission range for the long-range cruise (LRC), the maximum range cruise (MRC) and the high-speed cruise (HRC). For the LRC the energy per pax nmi shows a minimum at 6000 nmi, but between 4000 nmi and the design range, only a small difference in the same metric can be observed. For the MRC and the HRC, the difference in performance between the LRC is minimising with range.

In Table 4-31, extended results are presented for the design range as well as the 3000 nmi mission.

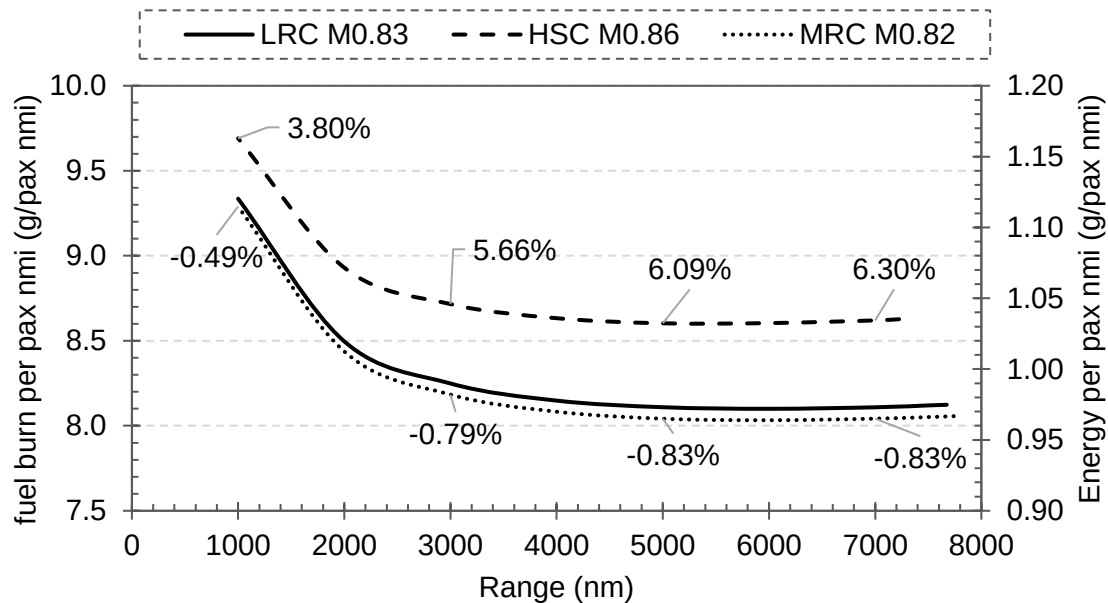


Figure 4-44 Off-design mission performance for typical payload

Table 4-31 Typical payload mission for the design range and 3000 nmi

	Units		
Max PAX Design Range	nmi / km	7674 / 14212	3000 / 5556
Max PAX Design Payload	t	47.61	47.61
Block Fuel (ISA)	t	25.81	10.24
Block Fuel per PAX. nmi	g/ PAX nmi	8.124	8.249
Block Energy	GJ	3097	1229
Block Energy per PAX.km	MJ/ PAX nmi	0.9748	0.9899

Finally to provide a better perspective on the scaled aircraft, EH2 BWB, a comparison is provided in Table 4-32 with the N3-X and the Boeing 777-9X. The data presented for the N3-X are derived from [28] and for the 777-9X from [109, 143]. For the later an estimation of the maximum payload was calculated for 160 kg/pax and the OWE was subsequently calculated according to the MTOW, MLW and the maximum payload. It is noted that for both the 777-9X and the N3-X the design mission is based on the maximum payload, whereas for the hydrogen-powered EH2 BWB the design mission is for the maximum fuel

capacity and a typical payload. In addition to this, by considering the different assumptions used for the sizing of the N3-X, definitive conclusions can not be drawn from this comparison. Nevertheless it is evident that for the LH₂-powered aircraft as the fuel becomes lighter compared to Jet A-1, the ratio between OWE and MTOW increases. Between the BWB designs as larger volumes are required to accommodate the hydrogen tanks, the longer fuselage together with the added tank weight results in a heavier OWE for the LH₂ application. A significant reduction in MTOW is observed between the T&W Jet A-1 and the BWB LH₂ aircraft which is primarily attributed to reduction in fuel weight due to the fuel properties as well as higher aerodynamic and propulsion system efficiency.

Table 4-32 Design characteristics for the Boeing 777-9X, N3-X and EH2 BWB

Aircraft	-	777-9X	N3-X	EH2 BWB
Airframe design	-	T&W	BWB	BWB
Fuel	-	Jet A-1	Jet A-1	LH ₂
EIS	-	2025	2035	2050
Range	nmi	7285	7500	7674
Number of Passengers	-	414	300	414
MTOW	t	351.5	214.8	246.8
MLW	t	266.3	179.2	240.9
OWE	t	<u>187.7</u>	121.3	170.9
OWE / MTOW	-	53%	56%	69%
Payload	t	<u>66.2</u>	53.6	47.6
Block fuel (Jet A-1)	t	86.2	39.9	25.8
Block Energy	GJ	3716	1721	3096
Block Fuel per PAX. nmi	g/ PAX nmi	28.57	17.73	8.12
Block Energy per PAX.km	MJ/ PAX nmi	1.232	0.765	0.974
Fuselage length	m	76.7	42	50.1
Wing span (with folding wingtips)	m	64.8 (71.8)	65 (-)	65 (-)

5 Conclusions & Future Work

This chapter summarises some key conclusions drawn throughout the course of this study as well as recommendations for future work.

5.1 Conclusions and discussion

On the LH₂ fuelled BWB:

- The BWB airframe appears to be a suitable architecture for LH₂ tank integration. On T&W configurations the tanks generally penalise the lift to drag ratio characteristics of the aircraft due to the added weight and surface areas. On BWB architectures, as the fuselage provides lift, extending it to accommodate the tanks provides a small counterbalance to that penalty.
- If the aircraft were to be designed for Jet A-1, then based on the cabin sizing requirements a linear stretch of +11.5% compared to the N3-X would be required. For the LH₂-fuelled version, the equivalent number is +17.5% which compared to the Jet A-1 scaled version instigates an increase of only +5.6%. As the two passenger cabins are sized exactly the same, they occupy exactly the same volume and the maximum diameter of the tank that can be fitted underneath the passenger cabin increases more than the linear factor.

On LH₂ tanks:

- Part of the contribution of this work was to investigate the benefits and the effect on tank sizing of pressure control strategies. Through the application of the dual-phase extraction, it was demonstrated that the tanks can become lighter and more compact (see Figure 3-20 and Table 3-7). In the case where both GH₂ and LH₂ are used for the propulsion system feed, the benefit maximises as no fuel is wasted and at the same time the tank does not have to be oversized to compensate for the GH₂ extracted. For the application examined, between tanks designed to use GH₂ extraction and the ones that do not, the former are overall approximately 40% lighter and can store 18% more fuel (Table 4-8).

Despite the difficulties of implementing a dual-phase engine feed system, such an application may allow for significant benefits.

- An additional contribution to knowledge was that for the first time the amount of additional LH₂ that needs to be carried, in order to compensate for its evaporation throughout the flight, was included as part of the tank sizing process (see equation (3-22)) and quantified. This was considered unusable fuel as it remains in the tank at the end of the mission as GH₂ and occupies the remaining tank volume. Relative to the useful LH₂ stored this was estimated to be a substantial quantity, in the proximity of 3% in terms of mass (Table 4-9 and Table 4-10).
- GH₂ temperature stratification may cause the pressure to rise faster than it would if both LH₂ and GH₂ had uniform temperature throughout the tank volume. Mixing the two phases would potentially eliminate this problem. In the context of this study, it is assumed that fuel sloshing during the accelerating or decelerating phases of the flight may achieve just that by providing forced tank content mixing and reduce the pressure. If fuel sloshing needs to be prevented, to minimise the potential travel of the CoG during the flight, other mechanisms for mixing need to be considered.
- LH₂ Tanks can adequately be designed without the need for GH₂ extraction.
- In case a mission profile dictates so, or the insulation is overengineered, relatively low heat leak in combination with the engine fuel flow demand may lead to pressure decrease. To avoid the pressure to drop below the ambient a heater may need to be considered.
- During descent and after landing, the heat input in combination with low LH₂ levels would cause the highest pressure rise rate throughout the mission. Using only GH₂ in the engines during descent would potentially rapidly decrease the pressure in the tank and allow for bigger margins after landing.
- During the ground hold before take-off, the high density of the hydrogen mixture and the heat addition to the tank would cause the LH₂ to expand

and an almost negligible quantity of GH_2 to liquefy. This might not be a significant problem for a limited period, however, in the examined case of the reserve tank (section 4.3), where no hydrogen was removed for at least 32 hours, LH_2 would expand and occupy almost all the inner tank volume. To account for this effect this tank was designed with an increased GH_2 ullage of 8% compared to the 3% considered in the other tanks.

On the propulsion system:

- The design space exploration for both TeDP configurations showcased the impact of selecting the LH_2 propulsion system design parameters based on the design mission performance. More specifically throughout the test cases presented, regardless of the number of distributed fans, compact designs allowed for the longest ranges (see Figure 4-24). For the turboshaft TeDP for example, FPR_{BLI} of 1.4 and 1.5 (at ToC) compared to 1.2 (which provided the lowest SFC, see Figure 4-16) provided 2-3% higher SFC but achieved 5-7% longer range as the aircraft became 11-13% lighter due to propulsion system weight savings. Exactly the same trend was observed when sizing the turbofan in the turbofan TeDP case.
- In addition, it was demonstrated that although larger numbers of fans may be a more efficient option, width limits associated with their installation prevented that. In fact, in the case of 12+ fans, the optimum FPR_{BLI} exceeded the maximum width limit set.
- The impact of VAN on the overall performance of the propulsion system was compared against designs with fixed nozzle area and their differences were quantified for the case presented in 12-Fan with fixed nozzle area. VAN allowed for higher TETs and OPRs which in turn enabled lower SFC and weight. These benefits were maximised for low FPR_{BLI} allowing for up to 3.8% longer range and were significantly

reduced for higher FPR_{BLI} . It is noted however that the weight of the VAN mechanism was not considered in this study.

- For the turbofan TeDP, this study addressed in detail for the first time the effect of the turbofan design on the mission level fuel efficiency. As the BLI fans are assessed based on installed performance, installation effects on the turbofan were also considered. The thrust split between the turbofan and the BLI fans was investigated and it was determined that in general the more thrust produced by the BLI fans the more efficient and lighter the system becomes, mainly because the BLI benefits are maximised (as shown in Figure 4-31 and Figure 4-32). It is noted, however, that the optimum thrust split may vary in case different assumptions are considered for the installation effects, the intake pressure losses and the transmission efficiencies of both the BLI fans and the engine fan.
- The turbofan TeDP effectively covered the design space of the turboshaft TeDP which was not feasible due to the BLI array width restriction. In most of the cases for varying numbers of fans and FPR_{BLI} , the turbofan was sized to provide the remaining thrust of the requirements imposed. The optimum configuration occurred at very low thrust splits, where the turbofan produced approximately 10% of the total thrust (5% from the BP and 5% from the core). This thrust requirement in a combination with the FPR_{TF} resulted in a small enough fan diameter and consequently high rotational speed that matched that of the LPT. This eliminated the need for a gearbox and the associated transmission efficiency losses as well as added weight and allowed for a direct drive between the LPT and the engine fan.
- The multipoint design approach implemented on TeDP showed that varying TET and OPR based on set maximum limits changes the evolution of the efficiency of the propulsion system against different design FPR options. As shown in the example of Figure 4-16, between the single point approach (where reasonable TET and OPR values are selected) and the multipoint approach, the latter corrected the TET and

OPR and dictated lower and higher SFC for lower and higher FPR_{BLI} respectively.

- The BLI fan array performance can be represented by a single fan installed at half the available semi span and then the power and thrust may be multiplied by the number of fans considered. The chord length used to calculate the BLI properties should reflect the actual length of that position. Its performance is with good approximation the average performance of the most outer and inner fans. It is noted, however, that this conclusion only applies to the current configuration with its listed assumptions.
- Designing for synchronous transmission between the LPT and the BLI fans coupled with VAN demonstrated its feasibility, at least theoretically. The pair pole ratio between the generator and the motors, even when discrete values are used, can efficiently couple the near-optimum rotational speeds of the two components. On the same aspect, no clear pattern was observed as to whether the LPT or the BLI fan should impose the rotational speed selection of the system.
- The weight of the electrical transmission system, although estimated with relatively ambitious high power to weight ratios, accounts for a significant proportion of the overall weight of the propulsion system (20-40% for the turboshaft case).
- In the context of this work, the selection of the optimum propulsion system was based on mission fuel economy alone. However, it was observed that a range of design options (number of fans, FPR_{BLI} , thrust split between BLI array and turbofan as well as FPR_{TF}) were within 1% of the maximum range achieved which translates to 77 nmi. When considering the development, manufacturing and maintenance costs of such complex systems this difference might not be a decisive factor for both manufacturers and operators.

5.2 Future work

Improvements to the framework:

- The aircraft performance can be expanded to account for CoG travel throughout a mission. Given this capability, the integration, positioning and sizing of the LH₂ tanks can be thoroughly addressed to account for that aspect. The same can be performed for the positioning of the engines, BLI fans as well as airframe components such as wings and potential installation of canards.
- On the LH₂ tank sizing, accounting for the GH₂ temperature stratification of the GH₂ can further be improved with higher fidelity methods. Dataset acquired through experiments or CFD may be used to correlate the stratification factor used in this work with the fill level of LH₂, the heat flux as well the geometric characteristics of the tank. Instead of the stratification factor approach, the methodology introduced by Ghafri et al. [151] (published at the time of the writing) can be used which shows promising accounting of the stratification effects.
- The propulsion system design space exploration for both cases was based on fixed thrust requirements for both ToC and EoR. These were determined according to the mean value of FPR of the examined range and were considered to be representative as they allowed for adequate mission performance for all the design space. An immediate next step would be to correct the thrust requirements based on the MTOW estimated, which requires several iterations between aircraft weight estimation, aircraft performance and propulsion system sizing. By doing so it is expected that FPR designs below 1.4 would require slightly higher thrust levels which would make them slightly heavier and further penalise the design mission range. For FPR above 1.4 exactly the opposite is expected.
- In this study, the thrust of the core was selected based on minimum ToC SFC. An improved approach would be to include the mid-cruise performance on the propulsion system sizing framework and reassess

the thrust split between the core and the BLI fans based on minimum cruise SFC.

- For the turbomachinery gas path sizing and weight estimation, common geometry inputs were used throughout the design space. In further iterations of the design, these inputs can be better refined to optimally reduce the weight of every examined design, especially the ones that dictate the geometry of the fan and the LPT (i.e. aspect ratio of rotor and stator blades, blade space to chord ratio etc.).

Recommendations for future work:

- Follow up on the last improvement would be to establish correlations between thrust requirements and the MTOW (or another more suitable metric). So far the ones that have been used, like the ones summarised in [150], refer to kerosene-fuelled aircraft. Updating those for LH₂-fuelled would be very beneficial for the preliminary design phases.
- Based on the aircraft model presented in the results chapter, perform a design space exploration for geared Turbofan and compare it with the TeDP configurations discussed. To fairly compare the technologies, however, the sizing process needs to account for the following:
 - installation effects (nacelle and pylon drag).
 - the gearbox model should be upgraded to account for varying transmission efficiency at off-design conditions.
 - similar to the electrical transmission, the heat losses of the gearbox need to be included in the LH₂ preheating system.
- An interesting extension to this study would also be to investigate the scaling of the application for different ranges and payloads and showcase how the propulsion system design space exploration changes depending on the size.
- An equivalent Jet A-1 powered BWB can be created and the difference in optimum propulsion system parameter selection can be demonstrated. A comparison between the LH₂ and Jet A-1 aircraft variants in terms of

energy consumption (similarly to previous LH₂ concept aircraft studies [20, 22, 30]) would also be beneficial.

- Techno-economic risk assessment and environmental impact studies of the LH₂ aircraft which would include estimations of the life cycle CO₂, mission level NO_x and contrail emissions. This could be extended to investigate mitigation strategies that would involve trajectory optimisation and/or changes to the aircraft and propulsion system optimum sizing.
- Heat management studies; previous studies and this one included have demonstrated that LH₂ preheating can be a challenge. Hydrogen due to its properties requires a significant amount of heat transfer to raise its temperature. In the context of this work, this was calculated as approximately 5% of the combustion heat release. Heat management system sizing studies could provide a very useful insight into the challenges associated with the pre-heating, and can determine the weight of the fuel transmission system as well as its power requirement. In addition, as shown with this application, the heat management system can synergistically be used to improve propulsion system efficiency with compressor intercooling, precooling, turbine, engine oil and environmental control system cooling.

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Appendices

Appendix A Methodology

A.1 Tank sizing

Figure A-1, shows more information on the tank performance based on the example and inputs discussed in section 3.3.4.

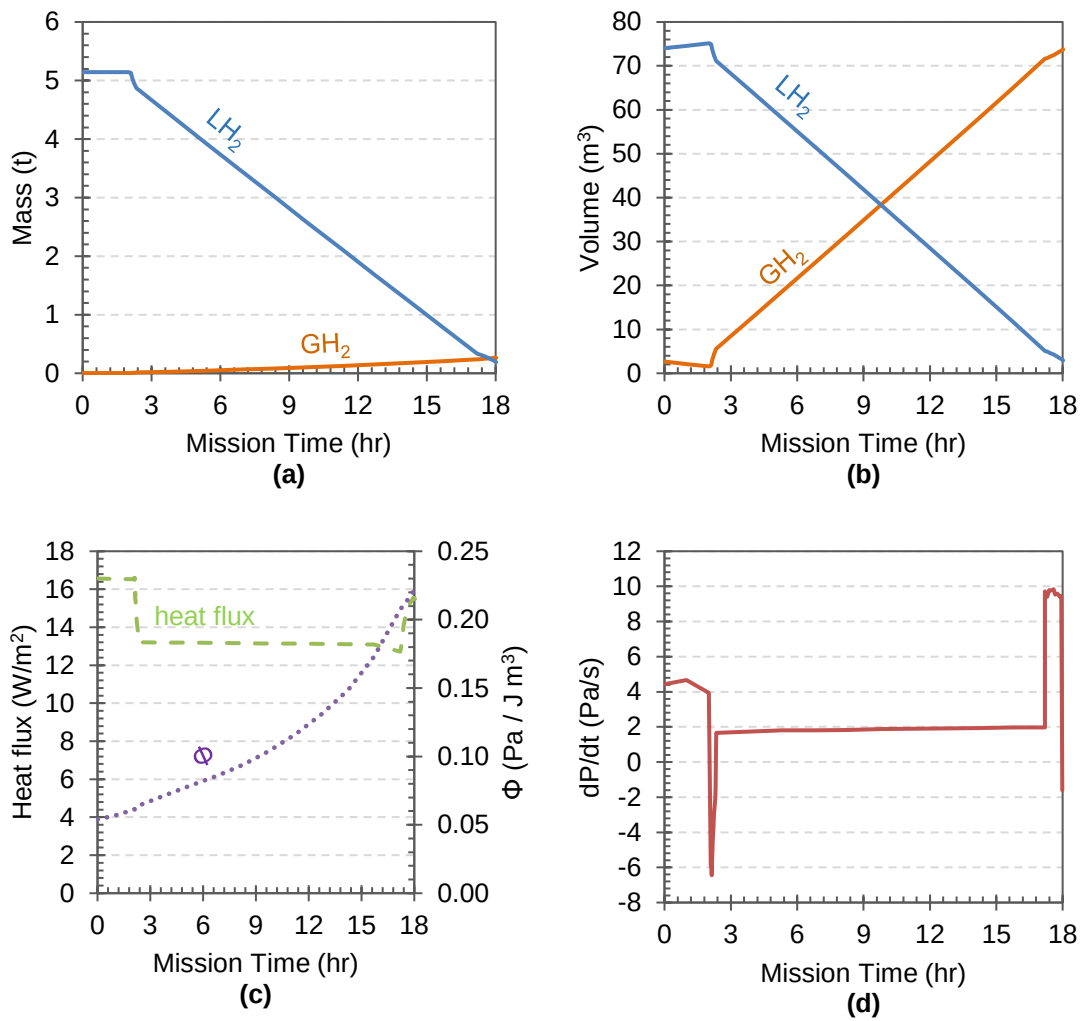


Figure A-1 (a) mass of LH₂ and GH₂, (b) Volume of LH₂ and GH₂, (c) heat flux and (d) rate of pressure change against mission time

Figure A-2 shows the η_{tank} evolution for different insulation materials and thickness based on the example of section 3.3.4

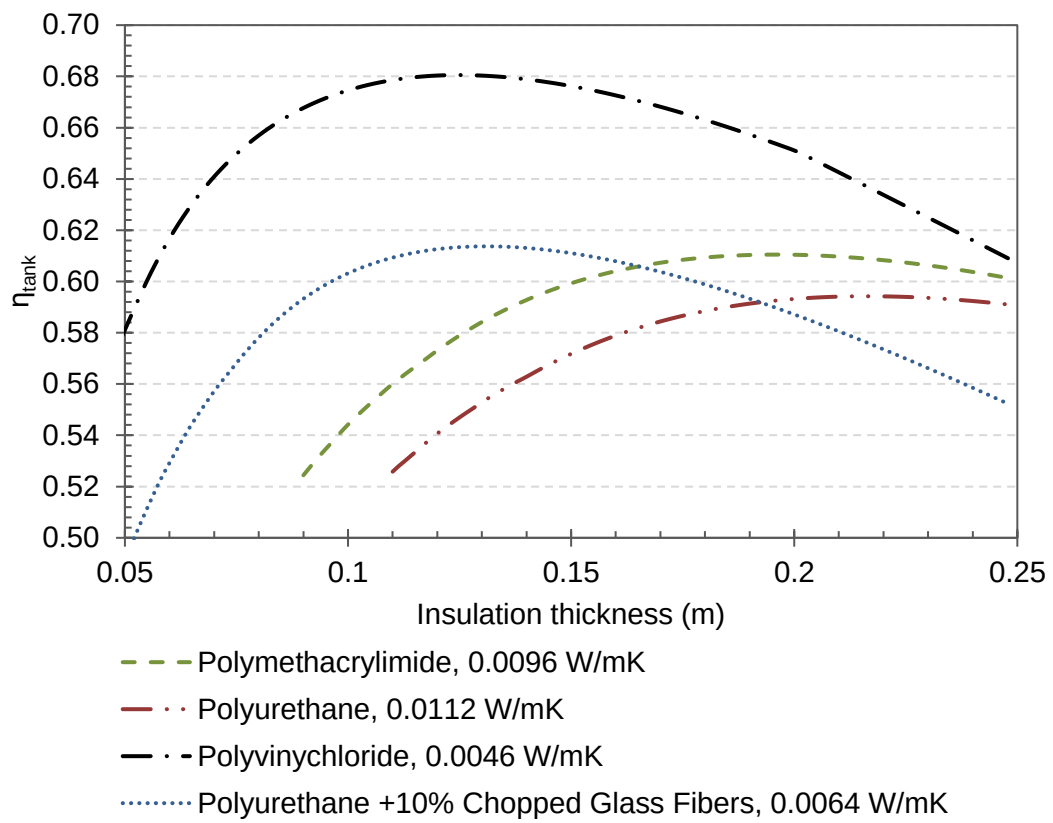


Figure A-2 Parametric study of the insulation thickness for different insulation materials [96]

A.2 BLI Propulsor

BLI Fan sizing inputs and results for the example of section 3.4.1.2.

Table A-1 VAN example key inputs and results

	Parameter	Units	Values
Inputs	Free stream Mach	-	0.82
	Altitude	m	11000
	Mass Flow	kg/s	100
	Fan Pressure Ratio	-	1.4
	Fan Isentropic Efficiency	-	0.935
Results	NPF	kN	9.08
	Power Consumption	MW	2.65
	Nozzle Area	m ²	0.843
	Nozzle Pressure Ratio	-	2.13
	Nozzle Non-dimensional Mass Flow	kg/s	203.5

A.3 Electrical

As shown in Figure A-3, for an operating temperature of 25K, the current density (J) decreases as the magnetic field strength (B_g) increases. However, according to eq. (3-47), both high values of B_g (directly) and J (indirectly through K) result to lower V_R . Therefore B_g is selected such that it maximises the term $J*B_g$. For the example assessed, this is achieved for a B_g of 1.2T at an operating temperature of 25 K with the resulting J of 155 A/mm². Based on the values selected in Table 3-8 and according to eq. (3-51), the remnant flux density of the magnet (B_r) is selected to 1.3 T.

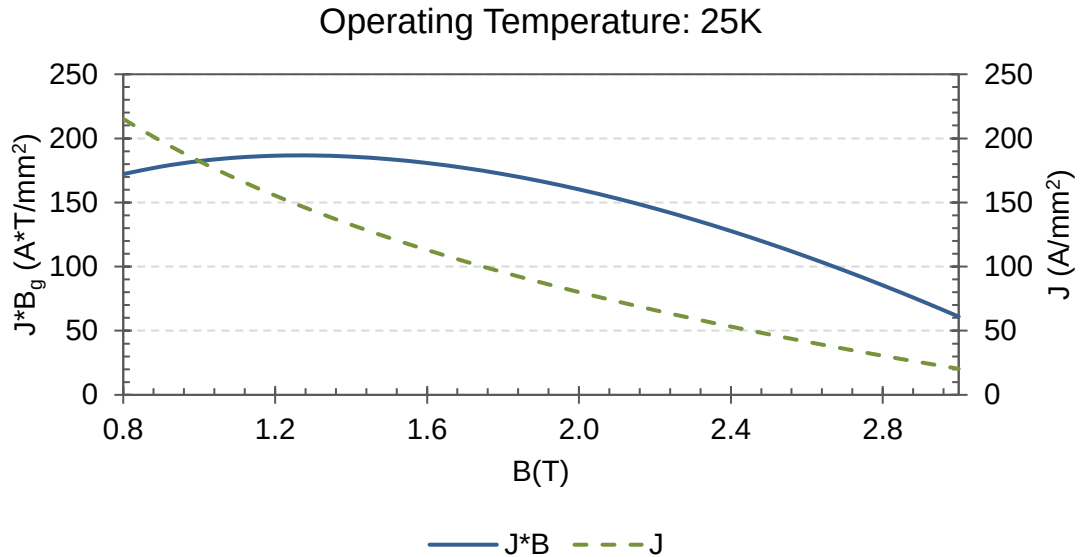


Figure A-3 Current density against the strength of the magnetic field

Appendix B Results

B.1 Cruise Mach number and altitude

To determine the most efficient altitude and Mach number for the design mission, a variety of cruise altitudes and Mach numbers are examined. For the assessment, an initial propulsion system model is created with two turboshaft engines powering 10 remote BLI fans. Figure B-1 shows the effect of cruise altitude and Mach number on the fuel mileage parameter. For every altitude, increasing the cruise speed initially increases but then rapidly decreases the range. Overall at 40000 ft the best performance is achieved and is selected for the design mission. Although the optimum speed is around 0.81 Mach, according to the aircraft requirements in Table 4-1, at least 0.82 needs to be selected. As a result, 0.82 Mach is set for the maximum range cruise (MRC) and 0.83 for the long-range cruise (LRC), which suggests a 1% penalty in range compared to the MRC.

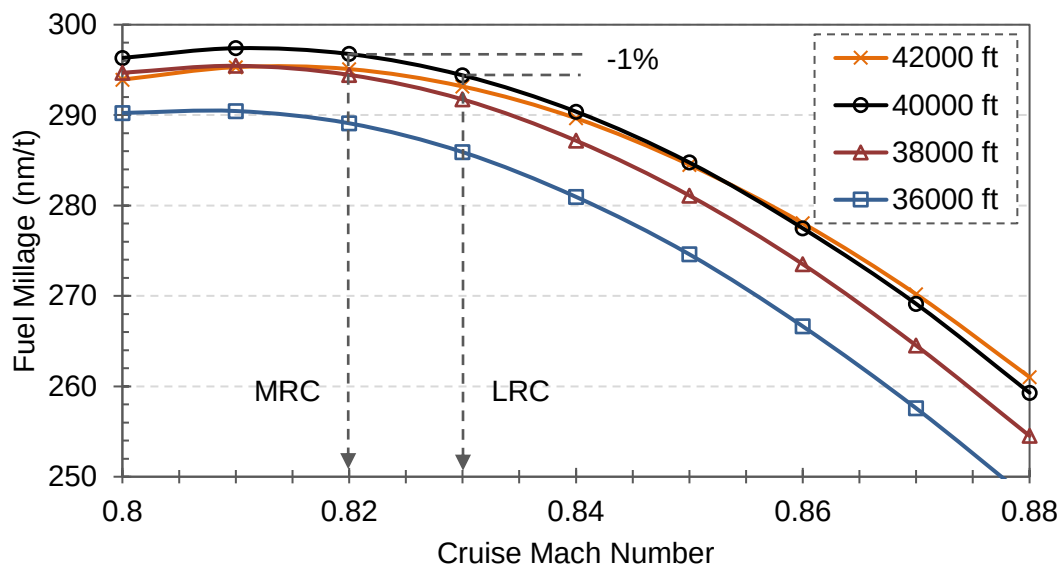


Figure B-1 Effect of altitude and Mach number of cruise

B.2 LH₂ tank results

Table B-1 and Table B-2 summarise the tank sizing results between the maximum η_{tank} designs for the cases with and without GH₂ extraction presented in 4.3.

Table B-1 LH₂ optimum tank sizing results with no GH₂ extraction

Parameter	Units	Values		
Tank numbering	-	1	2	3
External diameter	m	2.82	2.025	1.63
total length	m	24.35	22.51	15.22
length limit	m	24.38	22.71	15.55
Internal Volume	m ³	116.2	51.1	20.4
Total weight	kg	3360	2192	1188
Useful LH ₂	kg	7544	3292	1303
Gravimetric efficiency	-	67.7%	58.6%	51.0%
Mission vented GH ₂	kg	0	0	0
Mission evaporated LH ₂	kg	230.7	125.9	60.6
Maximum allowable pressure	Bar	1.69	2.50	3.44
Insulation thickness	m	0.14	0.14	0.14
Total LH ₂ stored	t	25.2		
Total useful fuel	t	24.3		
Total tank weight	t	13.5		

Table B-2 LH₂ optimum tank sizing results with GH₂ extraction

Parameter	Units	Values		
Tank numbering	-	1	2	3
External diameter	m	2.82	2.025	1.63
total length	m	24.09	21.96	14.55
length limit	m	24.38	22.71	15.55
Internal Volume	m ³	133.5	61.7	17.1
Total weight	kg	2285	1222	384

Parameter	Units	Values		
Tank numbering	-	1	2	3
Useful LH ₂	kg	8641	3977	1097
Gravimetric efficiency	-	76.9%	74.2%	71.7%
Mission vented GH ₂	kg	531.1	380.1	144.9
Mission evaporated LH ₂	kg	774.8	498.3	179.3
Maximum allowable pressure	Bar	1.38	1.50	1.62
Insulation thickness	m	0.045	0.045	0.045
Total LH ₂ stored	t	29.6		
Total useful fuel	t	27.4		
Total tank weight	t	7.78		

The gauge pressure evolution of the final selection of tanks is shown in Figure B-2.

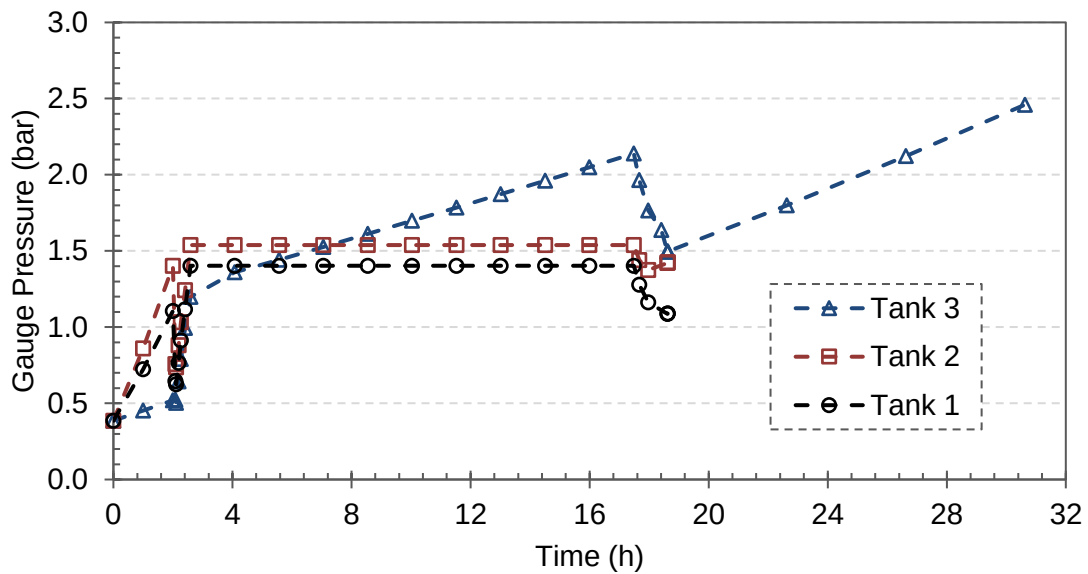


Figure B-2 Pressure evolution of the three tanks throughout the design mission

B.3 BLI power savings

In Figure B-3, the difference in power between a BLI and a free stream propulsor is presented based on the discussed sizing of section 4.4.2. It is noted that the following differences in sizing are considered:

- For the BLI fan: fan efficiency 93.5%, nacelle rectangular 1.15 and 2 times the diameter of the fan in width and length
- for the free-stream fan: fan efficiency 93.5%, nacelle cylindrical 1.15 and 2 times the diameter of the fan in diameter and length

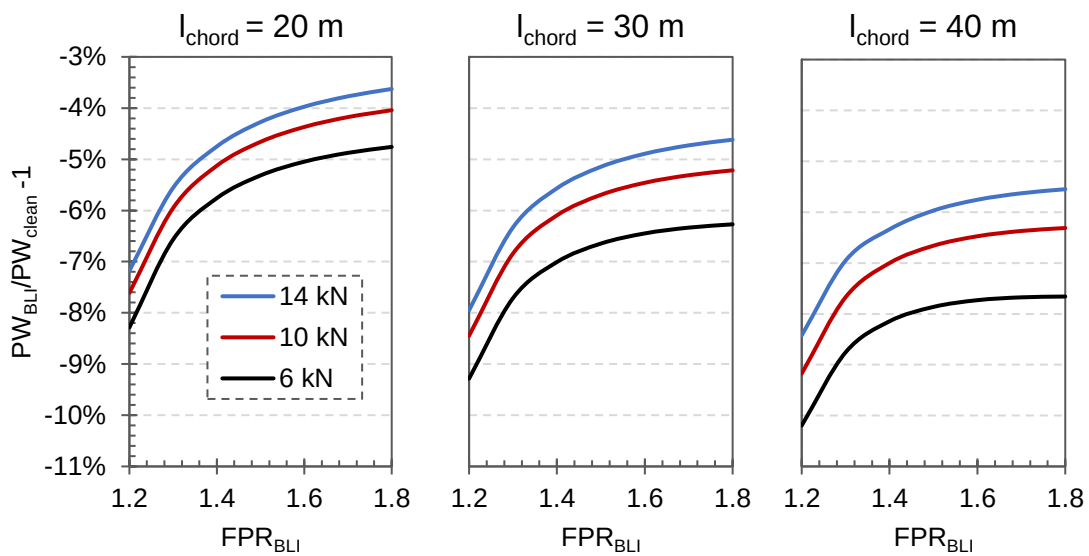


Figure B-3 Difference in power requirement between a BLI and a free-stream propulsor for varying FPR_{BLI} , NPF and l_{chord}

B.4 Turboshaft TeDP 12 fans fixed area nozzle

Subsidiary plots for the fixed nozzle area assessment of subsection 4.5.2.

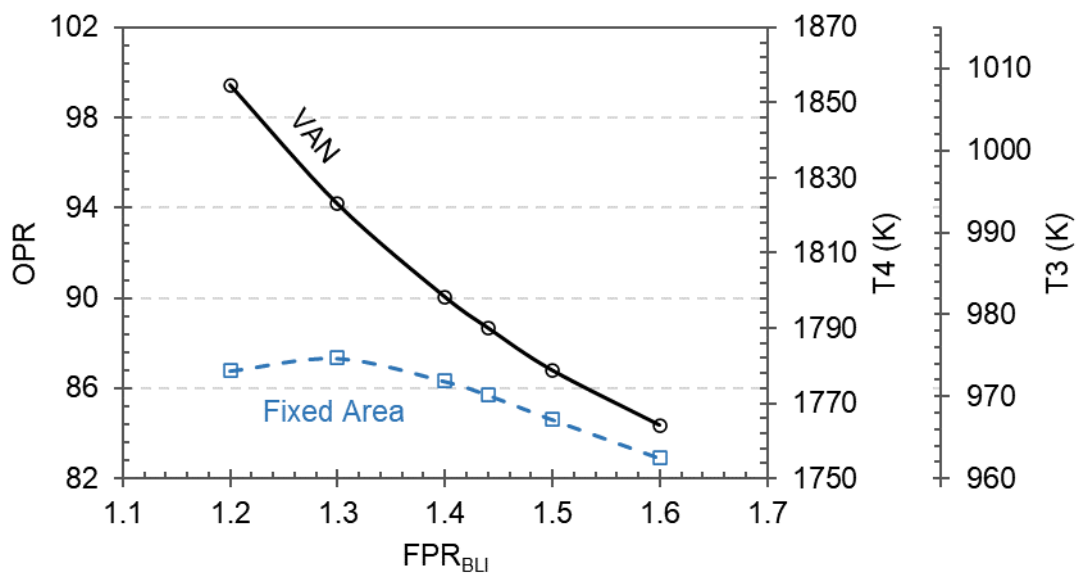


Figure B-4 DP OPR, TET and T3 variation over FPR_{BLI} after multipoint design for VAN and fixed area nozzle

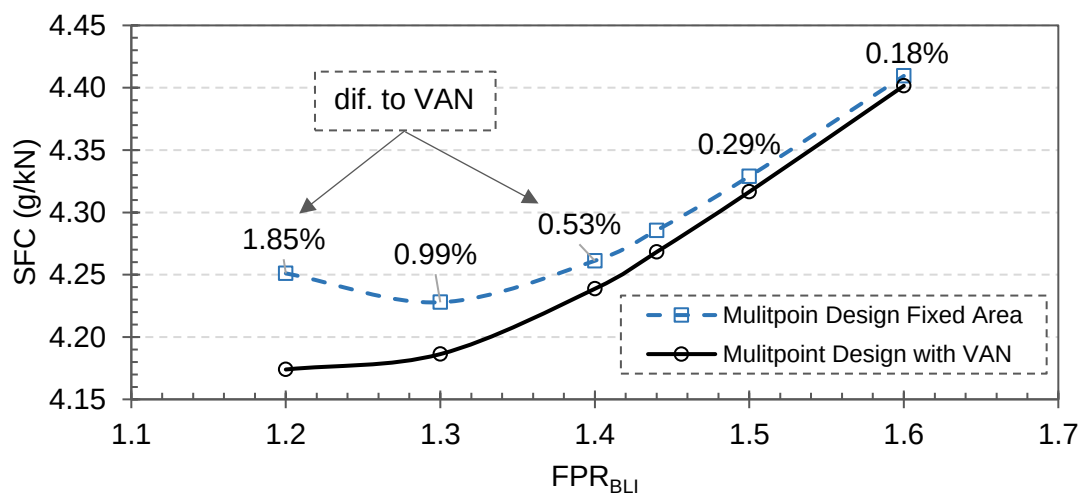


Figure B-5 SFC against FPR_{BLI} for multipoint sizing with fixed area nozzle and VAN

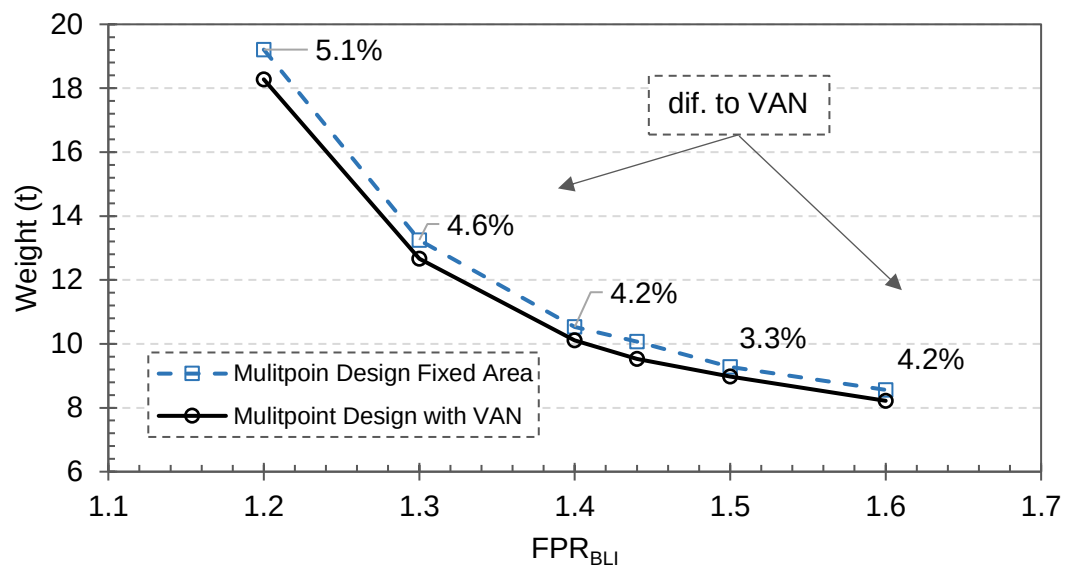


Figure B-6 Half propulsion system weight against FPR_{BLI} with fixed area nozzle and VAN

B.5 Turboshift TeDP 10-16 fans

and can be represented by Figure 4-15, Figure 4-17 and Figure 4-20 respectively

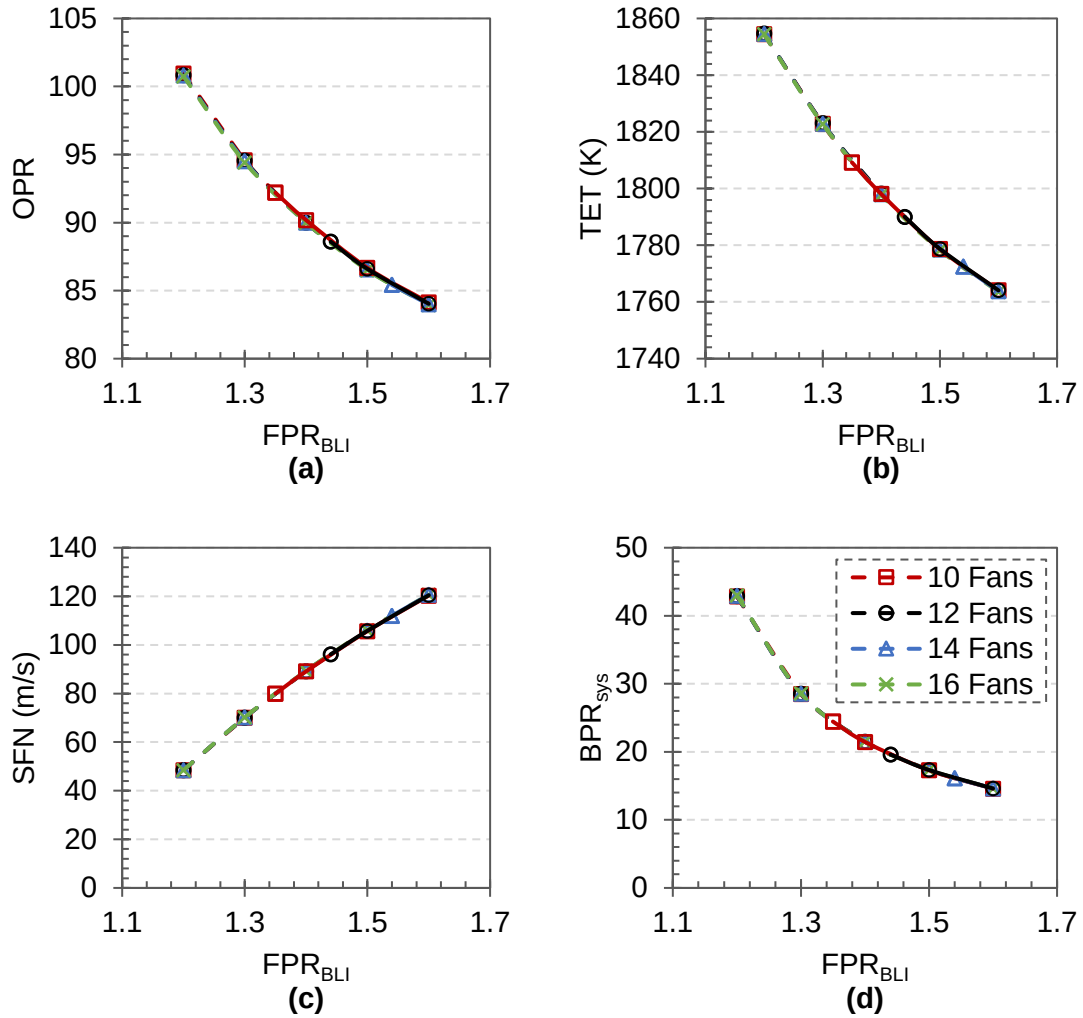


Figure B-7 (a) OPR, (b) TET, (c) SFN and (d) BPR_{sys} against FPR_{BLI} for different numbers of fans

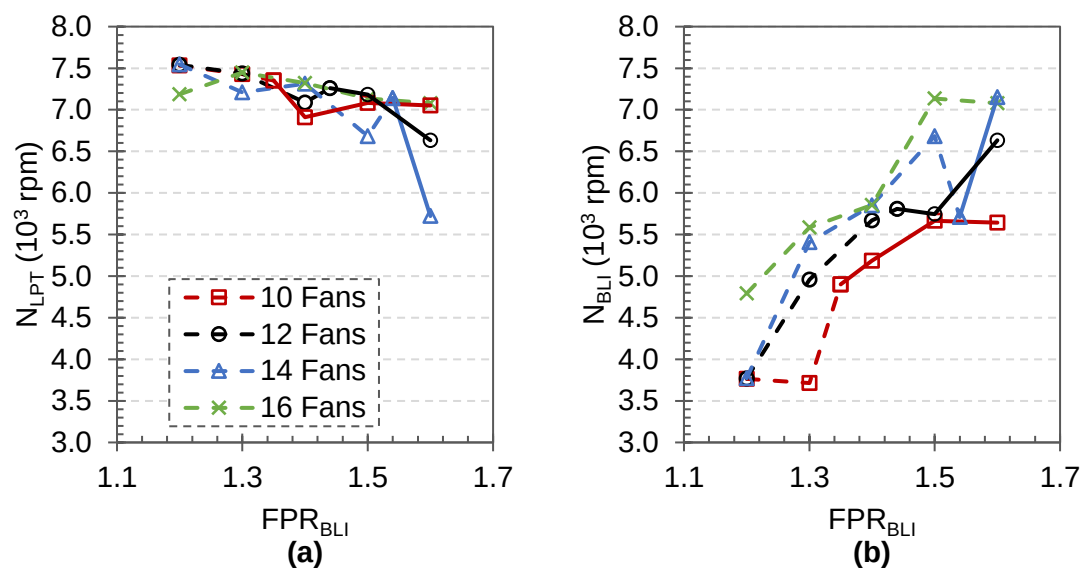


Figure B-8 (a) LPT and (b) BLI Fan rotational speeds against FPR_{BLI} for different numbers of fans

B.6 LH₂ aircraft results

Table B-3 Aircraft weight breakdown

Weight Build-up	Constituent kg	Part Frac
Structures	78659	31.9%
Wing Group	34273	13.9%
Horizontal Tail Group	0	0.0%
Vertical Tail Group	506	0.2%
Fuselage Group	43880	17.8%
Integrated Propulsion	34954	14.2%
Engines	3726	1.5%
Generators	2765	1.1%
Motors	3704	1.5%
BLI Fans	8184	3.3%
Nacelles	3669	1.5%
Fuel system & Tanks	12906	5.2%
Landing Gear	12500	5.1%
Main Landing Gear	11492	4.7%
Nose Landing Gear	1009	0.4%
Non-Propulsive	12703	5.1%
Flight Controls Group	2613	1.1%
Auxiliary Power Group	904	0.4%
Instruments Group	656	0.3%
Hydraulics Group	2535	1.0%
NPS/Secondary Electrical Group	1554	0.6%
Avionics Group	1701	0.7%
Environmental Control Group	2487	1.0%
Anti-Icing Group	254	0.1%
Final Man. Weight Empty Green	138815	56.2%
Furnishings and Equipment	32036	13.0%
Interiors/Outfitting Furnishing	24057	9.7%
Operating Items	7980	3.2%
Operational Weight Empty	170852	69.2%
Contingency	0	0.0%
Final Operational Weight Empty	170852	69.2%
Payload	47610	19.3%
Zero-Fuel Weight	218462	88.5%
Release Fuel (less Ramp Fuel)	28356	11.5%
Block Fuel	25781	10.4%
Reserves & Contingency	2575	1.0%
TOGW or MTOW	246818	100.0%