

CRANFIELD UNIVERSITY

ZIANG GAO

CONCEPTUAL DESIGN METHODOLOGIES APPROPRIATE TO
BLENDED WING BODY AIRCRAFT WITH BOUNDARY LAYER
INGESTION

SCHOOL OF AEROSPACE TRANSPORT AND MANUFACTURING
AIRCRAFT DESIGN

PhD

Academic Year: 2018 - 2021

Supervisor: Prof. Howard Smith
Associate Supervisor: Dr Amir Zare Shahneh
December 2021

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the degree of Ph.D.

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ABSTRACT

Blended wing body aircraft equipped with boundary layer ingestion technology represent a paradigm leap in aircraft design, one that provides significant aerodynamic and environmental benefits while reducing fuel consumption. Although there are numerous advantages to this structure, there are also various issues that arise as a result of the highly integrated nature of the configuration and the disciplinary couplings that ensue. This project develops a conceptual design methodology appropriate to the blended wing body commercial passenger transport with boundary layer ingestion and applied this methodology into the Cranfield multidisciplinary design analysis and optimisation environment – GENUS. The GENUS framework integrates a variety of aerodynamic analysis tools, an efficient geometry parameterization method, a semi-empirical mass breakdown model, and an effective boundary layer ingestion analysis model to enable the synthesis of a realistic conceptual design and exploration of the design space for this novel class of aircraft. The comparative case studies of blended wing body airplanes and conventional airplanes in three different sizes find that the blended wing body is more advantageous at larger aircraft sizes, which achieve 10.4% reduction in fuel consumption for 14000km-mission-range, 555-passenger class and the benefit could be expanded to 16.9% with the application of boundary layer ingestion. The parameter sensitivity analysis shows that the central body sweep is a key parameter for the design of the blended wing body aircraft, and the optimal central body sweep with the minimum fuel consumption increases as the cruise Mach number increases. In the final optimisation part, the geometry at Mach 0.74 has the lowest fuel consumption, but the one at Mach 0.86 has the highest productivity. The optimised geometry with boundary layer ingestion leads to a 10.8% reduction in the fuel consumption but an 0.8% higher structure weight compared to the optimised geometry without boundary layer ingestion.

Keywords:

Aircraft conceptual design, Multidisciplinary design optimisation, Blended wing body, Boundary layer ingestion

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LIST OF ABBREVIATIONS

AAA	Advanced Aircraft Analysis
ADP	Aerodynamic Design Point
AFC	Active Flow Control
AIP	Aerodynamic Interface Plane
AOA	Angle of Attack
AR	Aspect Ratio
BL	Boundary Layer
BLI	Boundary Layer Ingestion
BWB	Blended Wing Body
BPR	Bypass Ratio
CAR	Capture Area Ratio
CFD	Computational Fluid Dynamics
FFD	Free Form Deformation
FPR	Fan Pressure Ratio
FS	Free Stream
HAW	High Aspect Ratio Wing
KE	Kinetic Energy
KEG	Kinetic Energy Non-dimensional Group
MAG	Mass Flow Non-dimensional Group
MDO	Multidisciplinary Design Optimisation
MDAO	Multidisciplinary Design Analysis and Optimisation
MFR	Mass Flow Ratio
MOG	Momentum non-dimensional Group
MTOM	Minimum Take-off Mass
NPF	Net Propulsive Force
NASA	National Aeronautics and Space Administration
OEM	Operational Empty Mass
OPR	Overall Pressure Ratio
PC	Parallel Compressor
RANS	Reynolds-Averaged Navier-Stokes
TE	Trailing of the Edge of the Vehicle Aerofoil
UAV	Unmanned Aerial Vehicle

VTK Visualization Toolkit

LIST OF SYMBOLS

a	Speed of sound	m/s
A	Area	m ²
b	Span	m
$BAYW$	Passenger bay width	m
BPR	Bypass ratio	-
$\bar{c}$	Mean aerodynamic chord	m
c	Chord of the aerofoil	m
$CARGF$	Cargo aircraft floor factor	-
CPR	Compressor pressure ratio	-
C_d	Discharge coefficient	-
C_{Df}	Skin friction and form drag coefficient	-
C_{Di}	Induced drag coefficient	-
C_{Dtotal}	Total drag coefficient	-
C_{Dtrim}	Trim drag coefficient	-
C_{Dw}	Wave drag coefficient	-
C_F	Skin friction coefficient	-
C_{m_α}	Moment coefficient curve slope speed	-
C_L	Lift coefficient	-
C_{L_α}	Lift coefficient curve slope	-
C_p	Specific Heat Coefficient	-
d	Diameter	m
D	Drag	N
$DC60$	Distortion coefficient	-
D_{BLI}	Recovered drag by BLI	N
D_{eng}	Average diameter of each engine	m
D_i	Ingested aircraft drag	N
EPR	Engine pressure ratio	-
ETR	Engine temperature ratio	-
f	Fuel to air ratio	-
$f_{structure}$	Structure correction factor	-

F	Thrust	N
FF	Form factor	-
FPR	Fan pressure ratio	-
$Fuel_{frac}$	Fraction of fuel to total mass	-
F_N	Net thrust	N
F_{trans}	Laminar transition fraction	-
g	Gravitational acceleration	m/s ²
h	Height of the streamtube	m
h_{struc}	Height of structure	m
H	Altitude	m
$HYDPR$	Hydraulic system pressure	Pa
K	Total wake defect	-
$K_{Aerofoil}$	Korn factor	-
K_n	Longitudinal static margin	-
KE	Kinetic energy	J
l	Length	m
$\dot{m}$	Mass flow rate	kg/s
mom	Momentum flow rate	kg·m/s
M	Mass	kg
M_{yGG}	Moments refer the aircraft's CG	N·m
Ma	Mach number	-
Ma_{crit}	Local critical Mach number	-
Ma_{DD}	Drag divergence Mach number	-
Ma_{lim}	Maximum Mach number for engine	-
$NABR$	Number of seats abreast	-
n	Design normal acceleration factor	-
N_{closet}	Number of closets	-
N_{gally}	Number of galleys	-
N_{lava}	Number of lavatories	-
$NABR$	Number of seats abreast	-
$NBAY$	Number of passenger bays	-
NFE	Number of fuselage mounted engines	-
NPT	Number of tourist class passengers	-

NPB	Number of business class passengers	-
NPF	Number of first-class passengers	-
p	Static pressure	Pa
P	Total pressure	Pa
P_n	Pressure at start of atmosphere level n	Pa
$PITCH$	Seat pitch	m
PR	Pressure recovery	-
PW	Power	W
q	Dynamic head	-
Q_R	Fuel heating value	-
R	Specific gas constant of air	-
Re	Reynold number	-
$RSPCHD$	Percent chord of the HWB fuselage rear spar at the fuselage	-
$RSPSOB$	Percent chord of the HWB fuselage rear spar at the side of body	-
S_{aftb}	Area of the aft body	m ²
S_{flap}	Total movable wing surface area	m ²
S_{ref}	Reference area	m ²
S_{wet}	Wet area	m ²
SFC	Specific fuel consumption	g/kN/s
t	Static Temperature	K
t	Thickness of the aerofoil	m
T	Total temperature	K
T_n	Temperature at start of atmosphere level n	K
TET	Turbine Exit Temperature	K
TPR	Turbine pressure ratio	-
TR_{aftb}	Taper ratio of the aft body	-
$TSFC$	Thrust specific fuel consumption	g/kN/s
u	Air velocity	m/s
Δv	Differential volume	m ³
V	Velocity	m/s
V_D	Design dive speed	m/s

$VARSWP$	Wing variable sweep weight penalty factor	-
Vol_{fuel}	Total fuel volume	m^3
WF	Fuselage width	m
x	Chordwise distance from leading edge	m
X_{np}	Neutral point on x axis centre	m
X_{LP}	Total fuselage length	m
X_{LW}	Fixed length of side wall	m
X_{CG}	Centre of gravity position on x axis	m
XL	Total fuselage length	m
XLW	Fixed length of side wall	m
Y_{CG}	Centre of gravity position on y axis	m
Z_{CG}	Centre of gravity position on z axis	m
α	Angle of attack	rad
β	Twist angle	rad
β	Percentage of the wake which is recovered by BLI	-
γ	Specific heat capacity ratio	-
δ	Boundary layer thickness	m
δ_e	Trim angle	rad
δ_t	Ratio of the total pressure to the reference total pressure	-
δ^*	Displacement thickness	m
η	Efficiency	-
η_P	Propulsive efficiency	-
θ	Sector angle	rad
θ	Momentum thickness	m
θ_t	Ratio of the total temperature to the reference total temperature	-
θ^*	Energy thickness	m
λ	Wing taper ratio	-
λ_n	Temperature lapse rate	K/m
μ	Dynamic viscosity of air at altitude	Pa·s
μ_0	Dynamic viscosity of air at sea level	Pa·s

ν	Ratio of trailing edge kinetic energy defect which is re-energized to the whole defect.	-
ρ	Density	Kg/m ³
τ	Average thickness to chord ratio	-
ϕ	Perturbation velocity potential	-
ϕ_p^*	Wake dissipation factor	-
$\Delta\phi_j$	Jet interference force	N
Γ	Dihedral angle	rad
Λ	Sweep angle	rad

1 INTRODUCTION

1.1 Background of the Research

For years, the commercial aviation industry's constant attempts to improve aeroplane performance have been linked with a growing desire to minimise its environmental impact: reduced fuel consumption, reduced pollutant emissions (particularly CO₂ and NO_x), and reduced noise emissions. These objectives necessitate paradigm-shifting solutions, including new aircraft configurations, new technologies, and new flight methods. Among the numerous novel alternatives, one of the most promising is the Blended Wing Body (BWB) arrangement with Boundary Layer Ingestion (BLI) technology.

In comparison to the Tube and Wing configuration, the BWB concept exhibits superior aerodynamic performance; the lift-to-drag ratio would increase by approximately 15% for long-haul commercial transport missions requiring medium to large capacity (from 300 to 500 passengers). The overall take-off weight would fall by around 10% in terms of payload weight and range. These two benefits would result in a reduction in both fuel usage and CO₂ emissions. Additionally, the broad central body simplifies engine integration on the upper wing surface, decreasing noise emissions via masking effects and enabling more efficient BLI solutions.

Despite these compelling potentials, the BWB has not yet been converted into a commercial aircraft due to a number of obstacles to overcome. Stability and control limitations, propulsion-airframe-acoustics integration issues, and the complexities of attaining optimal trade-offs from conflicting design needs in a tightly connected aircraft configuration are just a few of the challenges to overcome. To address these issues effectively, appropriate BWB design procedures must take into account the coupled subsystems represented by a diverse set of disciplines and a large number of variables. Due to the complexity of this configuration, dedicated multidisciplinary design methods based on Multidisciplinary Design Analysis and Optimisation (MDAO) methodologies should be developed.

This research project was launched as part of the GENUS Project. The GENUS Project was initialized by Prof. Howard Smith, who has been working on aircraft design for many years and gains a great interest in new type air vehicle design. This GENUS Project aims to develop a multi-fidelity multidisciplinary aircraft conceptual design, analysis and optimisation environment for different kinds of aircraft. The BWB with BLI aircraft model being developed now is part of this project.

1.2 Aims and Objectives of the Research

The aim of the research is to develop a multiple variables design synthesis and optimisation tool for the conceptual design of the BWB commercial passenger transport with boundary layer ingestion based on the GENUS framework

According the overall aim, there are four objectives in the research:

1. To identify the appropriate design methodologies for BWB conceptual design. A robust packaging routine will be developed to solve the internal component arrangement problem for the BWB design. The design methodologies will be programmed into GENUS environment and make them run automatically and smoothly.
2. To incorporate BLI module into the conceptual design procedure. The BLI modelling method should be accurate, not consume enormous computation resources, and compatible with BWB design methodologies developed before.
3. To compare BWB aircraft with conventional aircraft in the GENUS environment. The comparison for different types of aircraft in the same design environment could eliminate errors caused by method selection, and thus can better quantify the advantages of BWB aircraft and BLI technology.
4. To explore the design space of the BWB aircraft configuration. Design parameter sensitivity analysis will be conducted to explore the influencing factors of the BWB aircraft design. Optimisation for BWB configuration will also be made at different cruise Mach number.

1.3 Structure of the Thesis

The thesis can be divided into four main parts. The first part (Chapter 2) presents a literature review on the blended wing body and boundary layer ingestion technology. Through previous research on BWB aircraft, the technical issues and challenges are identified. Due to the tightly coupled issues, a multidisciplinary design, analysis and optimisation environment is regarded as the essential approach to the design of the BWB aircraft. In terms of BLI, this section introduces the boundary layer ingestion theory and summarises previous research on this technology, which indicate previous research in literature sets the propulsion system as an independent research objective to explore the potential improvement and it is necessary to study aircraft and engines as a whole.

The second part (Chapter 3) introduces the BWB design methodologies inside a multidisciplinary design analysis and optimisation environment called GENUS. Methodologies appropriate to BWB are developed in the GENUS environment. The NASA SUGAR Ray is chosen as a model to validate the proposed method. The BLI propulsion system modelling workflow including boundary layer modelling, inlet characteristics modelling and parallel compressor modelling is integrated in GENUS.

The third part (Chapter 4) presents the application of the MDAO model for case study. BWB concept is compared to conventional tube and wing aircrafts in three different range categories to explore the benefit of this configuration. BLI is also introduced into this comparison to demonstrate its advantage.

The fourth part (Chapter 5 and Chapter 6) presents the results and conclusion of the design space exploration. Chapter 5 conduct the design parameters sensitivity analysis including geometry parameters and operational parameters to identify the most effective design variables for enhancing the performance of the aircraft. According to this analysis, the optimisation result for BWB baseline configuration in the different Mach number and BWB with BLI configuration are shown in Chapter 6.

Chapter 7 concludes with discussion on the methodology and results, conclusions and recommendations for future work.

2 LITERATURE REVIEW

2.1 Growth and Development in the Aviation Industry

Continuous development in civil aviation, combined with growing awareness of the impact of aircraft operations on the environment, has compelled the industry to work tirelessly to develop an alternative eco-friendly aircraft design. These efforts in aircraft design have not only met industrial need, but have also contributed to the development of an adequate propulsion system. According to Airbus studies, passenger aircraft air traffic will increase by 4.3% in 2032, as illustrated in Figure 2-1.

Historically, reducing fuel consumption has been a common goal of new technology development. In recent years, however, as global environmental awareness has grown, reducing the negative impact of aviation on the environment has emerged as a new main motivation in scientific study. Consequently, designers have an opportunity to develop a solution that can accommodate both the expected increase and the airworthiness standards linked to air and noise pollution, among other things. The BWB-BLI concept, for example, may be a solution that can meet these requirements.

BLI has been studied for use in conventional tube-and-wing aircraft concepts due to its low fuel consumption. According to research conducted by Bauhaus Luftfahrt e.V. [1], this arrangement can result in a 9.9% increase in efficiency over the estimated 2035 EIS aircraft baseline. NASA's STARC-ABL concept, as proposed by Welstead and Felder [2], has a similar layout and anticipates a 12% reduction in block fuel use during the design mission compared to a 2035 EIS aircraft baseline. It is not only possible to achieve low fuel consumption with the BWB arrangement, but the fan performance is also significantly improved due to the employment of turboelectric fans that are spread throughout the system. Known as the BWB, it is a unique aircraft concept design that features engines that are firmly incorporated into the aft area of the upper surface. Ingestion of a thick boundary layer is one of the benefits of this type of arrangement, which further reduces ram drag by reducing ingestion drag. The SAX-40 BWB,

developed by the Silent Aircraft Initiative, makes use of three boundary layer ingesting engine clusters [3], which swallow 16.6% of the airframe boundary layer [4].

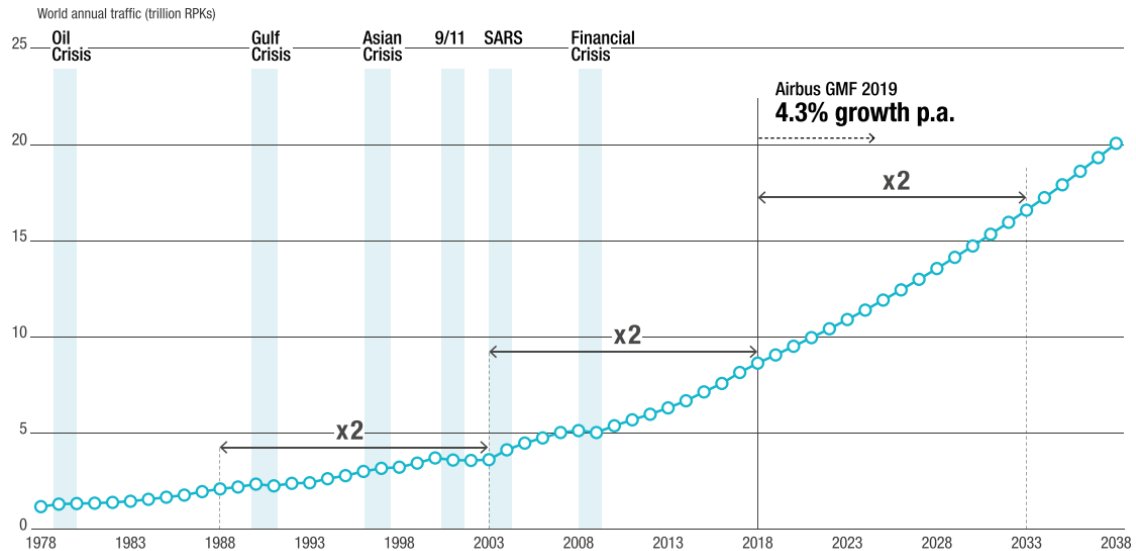


Figure 2-1 Air traffic growth for passenger aircraft [5]

According to the prognosis, the narrow and wide body aircraft markets can be supplanted by the BWB configuration. The BWB design features a lift-to-drag ratio (L/D) increase of 15–20% due to reduced drag caused by a 33% smaller wetted surface area than a typical tube and wing aircraft [6]. As a result of the above, the operational empty weight is reduced by 12% and the acoustic signature is reduced.

The impact of having a BLI concept for BWB configuration is not only to aid in the introduction of modern technologies, but also to aid in cost reduction, environmental impact reduction, and payload range expansion.

2.2 Blended Wing Body

2.2.1 Potentials and Challenges of the BWB Design

The tube-and-wing configuration with podded engines has historically been the dominating configuration for commercial aeroplanes. As a result, while major advancements to the airframe's performance have been made, modern aircraft appear to be very identical to their ancestors. One of the primary reasons for this

configuration's popularity is that a cylinder is an excellent pressure vessel, which is critical for an aeroplane that cruises at altitude. Additional enhancements to the conventional configuration are being investigated, such as strut bracing to accommodate large wingspans [7]. In addition, a 'double bubble' fuselage has been evaluated, which merges the cross-sections of two cylindrical fuselages into a single body [8]. Some of the new propulsion system concepts have been also explored for the conventional layouts. The STARC-ABL concept is a conventional design that incorporates a boundary layer ingesting technology [2]. This results in improved performance without significantly altering the aircraft's configuration. Additional performance gains, however, may be attainable by abandoning concepts derived from the conventional arrangement entirely.

The removal of the cylindrical pressure vessel opens the door to alternative designs. Liebeck [6] made an attempt to address a critical question: "Is there a renaissance for the long-haul transport?". His study's findings resulted in the development of a blended wing body idea, which is one such unique configuration. The BWB is a tailless aircraft design that integrates the wing and the fuselage into a single lifting surface, by the thickening of the wing in the central part of the airplane. This setup is advantageous in a number of ways. This concept reduces the wetted area to volume ratio, hence lowering the interference drag and increasing lift-to-drag ratio compared to conventional aircrafts. The BWB configuration also increases the volume of the cabin and could provide significant fuel savings (and hence lower pollution). The lifting-body fuselage also make an impact the aircraft's lift distribution favourably [9].

Liebeck et al. [10] concluded that a reduction of about 28% fuel burn per seat and a 15% reduction in take-off weight could be achieved for a BWB configuration with 800 passengers with 85 m span and a trapezoidal aspect ratio of 10. All in all, this could result in 10-15% lower direct operational costs. In 2000, NASA Dryden Flight Research Centre made an estimation of the potential benefits of the BWB configuration compared to conventional aircrafts for the year 2015, which predicts 12% lower Direct Operational Cost, 21% higher fuel efficiency, 6% lower gross weight and 17% lower emissions of NO_x brought by the BWB

configuration [11]. The SUGAR Ray hybrid wing body is another design being developed in support of NASA's N+3 goals. The arrangement reduces fuel consumption by 43.3% when compared to a reference 2008 EIS aircraft [7]. Noise reduction and safety against failure could also be improved if a number of engines are located above fuselage's huge upper surface. Thus, a BWB concept was used in the construction of the SAX-40 silent aircraft, resulting in a noise reduction of 28% when compared to a similarly sized conventional aircraft [12].

A blended wing body design presents a variety of issues. Most designs lack vertical surfaces, necessitating the use of multi-purpose control surfaces. Additionally, the airframe's stability characteristics may need fly-by-wire systems [10]. Due to the fact that the fuselage has a lifting body, the cabin angle may be greater than optimal during cruise [13]. Additionally, the high volume and greater distance from fuselage exits may complicate safe egress. This may also diminish passenger comfort, as the number of windows per passenger will presumably decrease.

The BWB configuration has the potential to further increase aerodynamic efficiency while facilitating the integration of other new technologies such as boundary layer ingestion and active flow control. However, the most difficult aspect of implementing such a configuration will be the implementation. The cost of establishing the necessary tools and facilities to build a significantly different airframe may be prohibitively expensive for a manufacturer. Similarly, the expense of developing new airport infrastructure that allows BWB aircraft to operate may be prohibitively expensive.

2.2.2 Recent Research Activity on the BWB

The aerospace industry and academia have increased their focus on specific areas of the development of civil commercial blended wing body configurations. Over the course of various research programmes, the aircraft configuration models employed for such activities have altered and evolved.

The McDonnell Douglas Corporation conceptualised the novel BWB design, which was recently proposed by the Boeing Company, as illustrated in Figure 2-2.

Its flying-wing configuration features a thick wing section, which maximises aerodynamics efficiency by combining the propulsion system, wing, and fuselage into a single lifting surface. This BWB concept also incorporates a large double-deck passenger compartment within the wing. There is substantial freight space adjacent to the passenger area. The BWB idea aims to develop an advanced long-range, ultra-high-capacity airliner capable of carrying 800 passengers in multi-classes at a cruise Mach number of 0.85 over a range of 8,000 nautical miles.

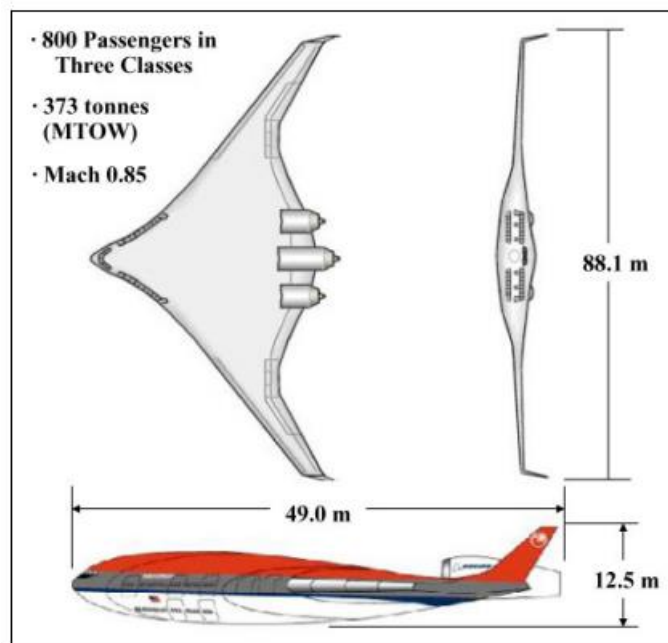


Figure 2-2 Boeing BWB concept design

The European Union (EU) has also launched a five-tiered programme for the development of innovative, efficient long-range, large-capacity passenger and freight aircraft [16]. This programme consisted of four entirely EU-funded framework programmes and one co-funded by the EU and the United States, as illustrated in Figure 2-3:

This aim of this research programme has primarily been to:

- Increase awareness of such unconventional configurations and their potential, particularly in the context of civil air transport.

- Identify risks and challenges associated with the BWB concept and propose solutions.
- Improve the design method and tools required to design and develop such configurations.

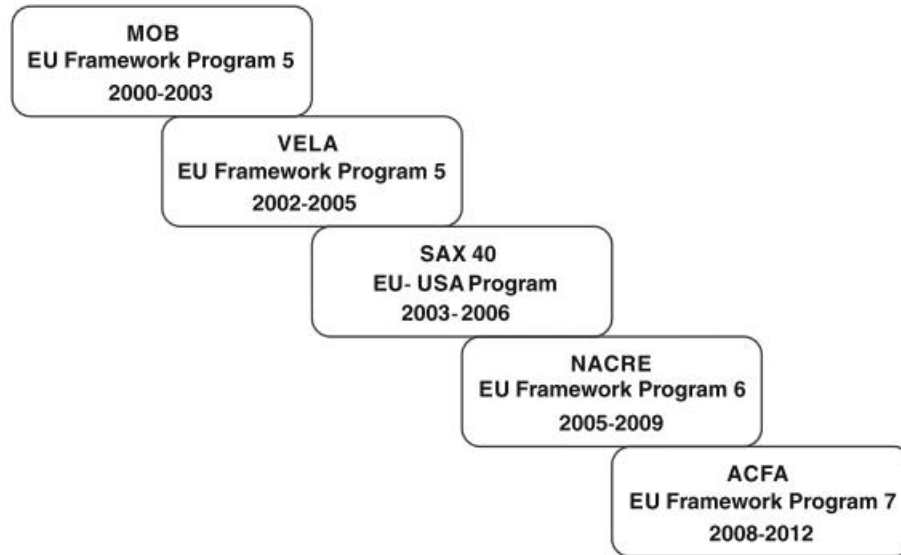


Figure 2-3 European Research Programs 2000-2012 [14]

Every research project in Figure 2-3 is devoted to the BWB aircraft design. MOB means Multidisciplinary Optimisation of a Blended Wing Body. The objective of the MOB project was to create a preliminary design model using a multidisciplinary approach. The MOB project has developed a so-called Computational Design Engine (CDE) to facilitate interdisciplinary optimisation within the MOB. The CDE encompassed four fields of study: Structural design, Aerodynamics and trim, Weight estimation and Flight mechanics.[15].

The Very Efficient Large Aircraft (VELA) project, shown in Figure 2-4, was designed on the concept of a very large aeroplane. The findings indicated that there were few issues with stability and control. It anticipated mass savings of roughly 10% and aerodynamic savings of 4–8%. Aerodynamic optimisation was mostly focused on the cruise period [16].

From 2005 to 2010, the NACRE setup (Figure 2-4) followed VELA. This project was established to integrate and validate technologies that enable the evaluation

of novel aircraft designs The emphasis was on environmental stewardship and passenger comfort [17].

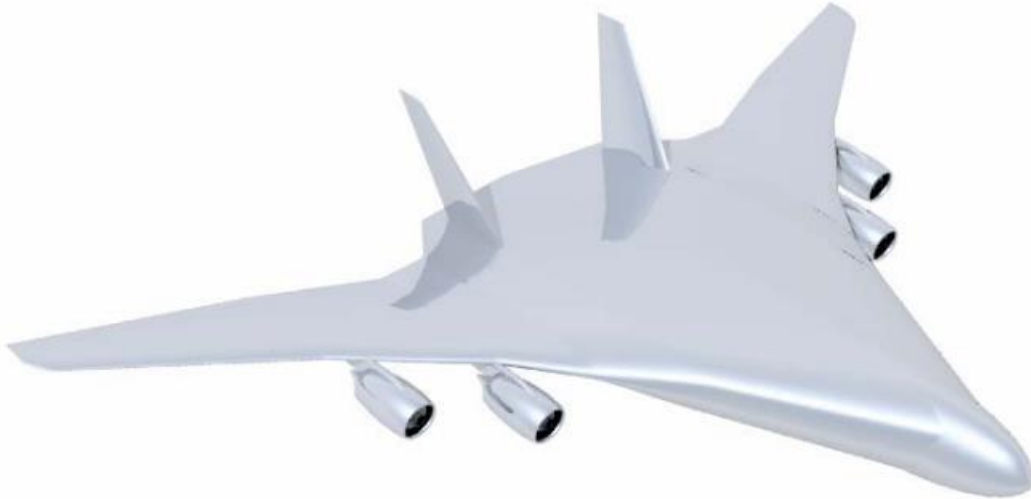


Figure 2-4 VELA BWB Configuration

The Silent Airplane Initiative, which began in 2003 as a collaborative European-American collaboration, aims to identify solutions to drastically reduce aircraft noise. BWB variants with embedded engines were among the technologies investigated. The SAX-40 (Figure 2-6) was one of the aircraft designs that resulted from this research [18].

From 2008 to 2012, the European Commission financed the Active Control for Flexible Aircraft (ACFA) 2020 project, which was primarily focused on the BWB configuration. The design holds 450 people and is powered by two podded turbofan engines (Figure 2-7). The engines are located aft on the wing's centre section. It focused on structural load reduction and ride comfort enhancement by actively regulating the aircraft throughout the mission with a effective, adaptive multi-channel control [19].



Figure 2-5 NACRE BWB Configuration

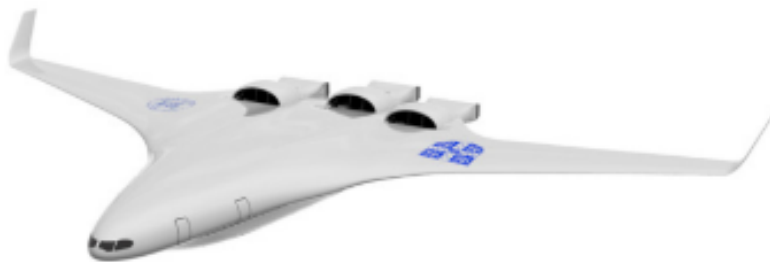


Figure 2-6 The SAX-40 BWB Configuration



Figure 2-7 The ACFA BWB Configuration

2.2.3 Current Methodologies for Aircraft Conceptual Design Modelling and Simulation

This section discusses the state-of-the-art design modelling approaches in aviation area that are not limited to BWB aircraft and can thus be used to any new design.

2.2.3.1 Overall Design

The conceptual design of an aircraft is the initial phase in developing a new aircraft. Beginning with a set of criteria, several configurations are investigated, from the sizing of major components such as the fuselage, wings, tails, and power plant to performance and cost analysis. The objective is to evaluate whether an aeroplane capable of meeting the requirements can be created in principle and, if so, what configuration is most suited.

A simple and time-saving procedure must be employed throughout the conceptual design phase, which involves the investigation of numerous combinations. The traditional research approach is based on a combination of empirical and simple analytical equations that are presented in aeroplane design manuals ([20]–[23]). Aircraft design is an iterative process that involves iterating across multiple disciplines such as propulsion, aerodynamics, structure and weight until a viable configuration is found that meets all requirements. Computers enabled the automation of analysis required for aircraft design. As an example, Advanced Aircraft Analysis (AAA) [24] is a software that can be used to size or re-engineer current aircraft. It is from Jan Roskam's "Airplane design" [21] and one of his publications on flight mechanics[25]. Raymer Design Software (RDS) is another example [26]. This aircraft design programme was developed in conjunction with the book "Aircraft Design: A Conceptual Approach" [20]. As with AAA, engineers can make modification to design variables in RDS after each iteration. In addition, RDS incorporates an optimisation method that allows the design parameters to be optimised. The outcome can be displayed graphically as a Pareto frontier.

The difficulty in aircraft design lies in determining the global optimum, as the design process involves many conflicting disciplines. A slight sacrifice in one area

may result in enormous benefits in another, resulting in an overall enhanced solution. Additionally, rising complexity and the interaction of technologies necessitate an integrated design and optimisation strategy that addresses the numerous disciplines concurrently. This makes it possible to discover globally optimal designs. To address these impediments, novel aircraft design techniques should be implemented. Multidisciplinary design optimisation is a relatively new discipline (MDO). The literature contains a concise summary of MDO approaches [27][28]. For example, this methodology is employed in a design process developed by ONERA [29].

Conventional configurations can be designed efficiently using the experience manual, but for some new configurations, the lack of a new configuration database requires a more general procedure to handle them. As processing capacity increases, more accurate performance predictions can be made for emerging technologies and aircraft configurations using methods based on fundamental physics. One such program is PrADO (Preliminary Aircraft Design optimisation), which was developed at the Technical University of Braunschweig [30]. It includes more than 50 analysis modules with a variety of various approaches for the design of aircraft with different requirements. Another example is a conceptual design process for small electric unmanned aerial vehicles created at the Technical University of Munich [31], which, like the previous example, requires a different method of aerodynamic analysis and weight estimation than larger aircraft flying at a high speed.

2.2.3.2 Propulsion

Propulsion performances and integration with the airframe are optimised by propulsion modelling. The ACFA project utilises the Gasturb Software [32] and a confidential general engine model. Reference [33] utilized the Gasturb programme in conjunction with an aircraft design framework to review the state of the art in propulsion modelling in his thesis: "The methodological contributions published in the United States of America, primarily used shared solutions for disciplinary analyses, such as the Flight Optimisation System (FLOPS) [34], the Weight Analysis of Turbine Engines (WATE) code [35], and the NASA Engine

Performance Program (NEPP) [36]." In 1992, the Integrated Propulsion/Airframe Analysis System (IPAS) [37] was utilized to connect these codes. IPAS has been validated in a high-speed aircraft design example [37], allowing for simultaneous optimisation of aircraft shape and engine parameters. COMETSBOARD [38], a design code integration platform, was presented as an optimisation engine for interdisciplinary aircraft and propulsion system design [39]. Geiselhardt also developed an optimisation method for aircraft design and its engine cycle for supersonic civil transport aircraft [40]. [41] provides an overview of notable propulsion system simulation software. [33] provides an overview of both proprietary ("in-house") and commercially available software solutions.

In Europe, the German Aerospace Centre (DLR) works not only on the integration of aircraft design frameworks, but also on "the integration of engine preliminary design into the overall aircraft design process via technology integration for the Virtual Aircraft (TIVA) [42]". TERA (Techno-economic and Environmental Risk Assessment) is a tool developed as part of the VITAL project and subsequently refined inside the NEWAC project [43]. It aims to provide a multidisciplinary assessment of the environmental impact and cost of ownership of engines [44]. TERA consists of several analysis modules, including aerodynamics, engine performance, noise, emissions and economics. Cranfield University's internal code TURBOMATCH [45] is developed to evaluate engine performance. It provides engine operational characteristics and a matching flow chart.

2.2.3.3 Aerodynamics

The main goal of any aerodynamic modelling effort is to determine the forces and moments generated by the aircraft. In general, modelling methodologies differ in their correctness because they perform permissible simplifications to reduce the complexity of the computation and thus the computation time. A high level of complexity does not always imply accurate results. Depending on the design, a large experimental database can yield equally satisfactory results [46][47]. The most significant differences among the various methods are flow characterization (laminar or turbulent), boundary layer considerations (viscous or inviscid) and flow velocity (subsonic, transonic, hypersonic), also classified as incompressible

or compressible flow. Only the most precise procedures will be capable of coping with all of the consequences of flow. Since high-accuracy methods require a lot of computational resources and preparation, some lower-accuracy but efficient methods are often used in the conceptual design process to obtain results quickly. The next section introduces commonly used computational methods for calculating an aircraft's aerodynamic forces and moments.

2.2.3.3.1 Low-fidelity Methods

The wing is a critical component for determining lift, drag, and moment characteristics. It is typically broken down into cross-sectional cuts for pressure distribution solutions, both mathematically and in use requiring only input of critical parameters into existing computational algorithms for calculation. This makes it possible to derive the two-dimensional forces and moments of the airfoil and then integrate them across the lateral direction along the entire span to provide a three-dimensional solution.

Preliminary estimations for the standard tube-and-wing aircraft can be derived from handbook procedures such as [20] and [48]. XFOIL [49] is a software programme for two-dimensional airfoils and XFLR5 is an upgraded version that incorporates calculation for three-dimensional wing design. The Panel method is also a common method for aircraft design. They subdivide the geometry into trapezoidal sections of different lengths and widths, from which they calculate the average pressure and transfer it to the next panel at the reference point until they integrate the pressure distribution over the entire span. The difference in the panel approach is whether the pressure is averaged or concentrated on the 25% line. These methods assume potential and incompressible flow [46]. This method is sometimes referred to as the vortex lattice method. For most applications involving small angles of attack and thin airfoils, panel splitting and force and moment calculations are sufficient. The user may choose between viscous and inviscid calculating methods. [50][51] and [52] are notable software examples. The first two use the vortex lattice method for linear aerodynamic wing design, while the latter allows the use of the vortex lattice method, lift line theory or the three-dimensional panel method.

The PAWAT program extends the lift line theory to take into account the effect of propeller-wing interactions [53]. The following part explains the operating principles of this software [53]: "This article's lifting-line approach is based on [54], which enables it to be utilised for systems of lifting surfaces with any camber, sweep, or dihedral. Additionally, by solving a nonlinear system of equations, the approach is capable of accounting for nonlinear airfoil data. The aerodynamics of a lifting surface are modelled discretely using a composite of horseshoe-shaped vortices that model the bound vorticity distribution over the wing's surface and the free vorticity distribution in the trailing vortex sheet. Each horseshoe vortex's bound section is aligned with the wing quarter chord line. The trailing vortices are either aligned with the free stream velocity or with the velocity of the local flow. The trailing vortices can be modelled to follow the wing surface (including flap deflection) and exit at the trailing edge."

The wing aerodynamics are superimposed using a slipstream and propeller model. "The propeller model is based on a blade element method, in which the propeller blades are separated into blade elements along their radial axis. Additionally, the propeller is represented using a variety of azimuthal blade orientations to accommodate for non-axial flow situations and periodic geometry changes such as blade flapping and pitching. Several models have been used to calculate the self-induced velocity at the propeller blades: (i) basic momentum theory as described in [55], (ii) blade element momentum theory as described in [56], and (iii) [57] to compute axial and tangential induced velocities. The lifting surface and propeller aerodynamics are connected by iteratively computing the system using the induced velocities" [53].

2.2.3.3.2 High-fidelity Methods

Typically, more precise methods are numerical in nature. They solve the flow domain around the aircraft very precisely at discrete points because the modelling methodology theoretically incorporates all the complexity of flow factors such as friction, turbulence and compressibility. The entire solution process involves mesh generation (usually associated with CAD systems), pre-processing (establishing flow conditions), and simulation and post-processing (extracting

forces and moments from the flow modelling). Despite significant advancements in research over the previous decade, the modelling process requires a significant amount of effort. Structural and aerodynamic impacts can be coupled. Aerodynamic theory is the theoretical basis of computational fluid dynamics (CFD) , which is described in detail in [58].

Diverse degrees of precision predominate. The Navier-Stokes equations provide the maximum accuracy. Although the mathematical formulation using the Navier-Stokes equation is more detailed physically, the simplified RANS equation is used more frequently because of the longer computational time required for direct numerical simulation (DNS). To solve a discrete flow control volume, a structured or unstructured mesh is usually constructed manually and then the flow physical field is solved at each mesh point using an appropriate solver. This requires significant processing resources, especially in the case of three-dimensional analysis. Recent studies have made extensive use of automated mesh creation and dynamic mesh adaptation, as described in [59].

CFD was formerly used in the later, more sophisticated stages of development. However, when dealing with complex flow phenomena or when test facilities are unavailable, its application in predesign becomes more prevalent. The proper trade-off between a fast and efficient but low-accuracy method and a time-consuming but accurate CFD simulation is becoming increasingly important for preliminary design. When performed by qualified methods, the analysis procedure frequently produces excellent lift and momentum forecasts that correlate well with test data.

Ansys, VS Aero, the TAU code and elsA are all well-known pieces of software in CFD simulation. VS Aero [60] has a simplified modelling environment to quickly create results. Comprehensive CFD software packages, including structural and thermal interaction, are available commercially from Ansys with its modules CFX and Fluent [61], as well as academically from the German Aerospace Centre (DLR) with TAU [62] and the French Aerospace Centre ONERA with elsA [63].

2.2.4 Optimisation in the Design of Blended Wing Body Aircraft

Optimisation technique has been extensively used during the BWB design process. Kuntawala [64] optimised the aerodynamic shape of a BWB with a high-fidelity inviscid Euler solver and adjoint-based gradient evaluation and sequential quadratic programming optimisation method. The optimisation objective was set as minimum amount of wave and induced drag on a BWB at transonic condition and the optimisation was conducted by altering the control points of the B-spline geometric parameterization of the BWB aerodynamic surface. The results shows a considerable reduction in drag compared to the baseline design.

Antoine and Kroo [65] used a Multi-disciplinary Optimisation (MDO) framework to investigate the viability of decreasing a commercial airline's operating costs while meeting noise and pollution limits. According to the study, the BWB design is the most environmentally friendly option due to its naturally low noise signature and minimal emissions.

The EU's MOB programme [15][66] used a response surface optimiser to undertake a multi-level optimisation on a BWB that combined aerodynamics, structural, and flight mechanics disciplines. This combination ensures that the maximum range of the BWB is achieved at constant take-off weight while adhering to stress level limits and controllability margin. However, this study was conducted in a distributed environment throughout Europe and does not take into account turbulence, packaging, propulsion, or other disciplines that may affect the design of a BWB.

Diedrich et al. [67] used the MDO technique in the SAI programme to develop a low-noise, aerodynamically efficient and economical BWB. Using WingMOD, a Boeing MDO code [68], this study lowered aircraft noise but fell short of the SAI aim. However, the study demonstrates the danger inherent in utilising noise as the objective in the MDO system. As a result, Leifsson et al. [69] recommend optimising performance while adhering to specific noise restrictions.

Ko [70] used the MDO technique to conceptually develop a BWB with distributed propulsion. Due to lower induced drag and enhanced propulsive efficiency, the

study discovered increased control efficacy and a 5.4 percent reduction in take-off gross weight. Additionally, these propulsion arrangements resulted in a 3% reduction in necessary thrust and a 7.8% reduction in fuel consumption. This is one of the most exhaustive studies yet conducted on the BWB synthesis. However, it does not take handling or ride quality into account when defining the problem.

2.3 Boundary Layer Ingestion Technology

2.3.1 BLI Theory

As indicated previously, this section's objective is to demonstrate the consequences of boundary layer ingestion on a blended wing body aircraft with the embedded propulsion system and S-shape duct. To acquire findings that demonstrate these effects, it is necessary to grasp the notion of wake ingestion and how it enables a reduction in fuel consumption while increasing propulsive efficiency. For the BLI system placed at the tail of the conventional tube-and-wing aircraft, its research is out of the scope of this study and will not be described in detail here.

2.3.1.1 BLI Aerodynamics

The operating principle of BLI may be understood from a propulsion standpoint by employing traditional propulsion control volume methods and one-dimensional averaged flow. According to Plas et al. [71], for the same aircraft and the same flight conditions, the power required to generate a specific net force from a BLI engine is less than the power required from a non-BLI engine, which can be summarised as follows:

Two idealized situations have been plotted in Figure 2-8: the first one is a podded engine without boundary layer ingestion and the second is an ideal boundary layer ingestion engine, which ingests 100% of wake. In cruising with a podded engine, the momentum deficit created by the aircraft total drag D_A must be balanced by the momentum surplus created by the engine thrust F_{engine} , using the following equation:

$$F_{engine} = \dot{m} \cdot (V_j - V_{in}) = \dot{m} \cdot (V_j - V_w) = D_A \quad (2-1)$$

where V_w is the wake velocity. The rate of mechanical energy, $P_{Non-BLI}$, given to the flow by the engine is:

$$P_{Non-BLI} = 0.5 \cdot \dot{m} \cdot (V_j^2 - V_{in}^2) = F \cdot \frac{V_j + V_0}{2} \quad (2-2)$$

the useful power P_{useful} required for flight can be obtained similarly:

$$P_{useful} = D_A \cdot V_0 = \dot{m} \cdot (V_j - V_0) \cdot V_0 \quad (2-3)$$

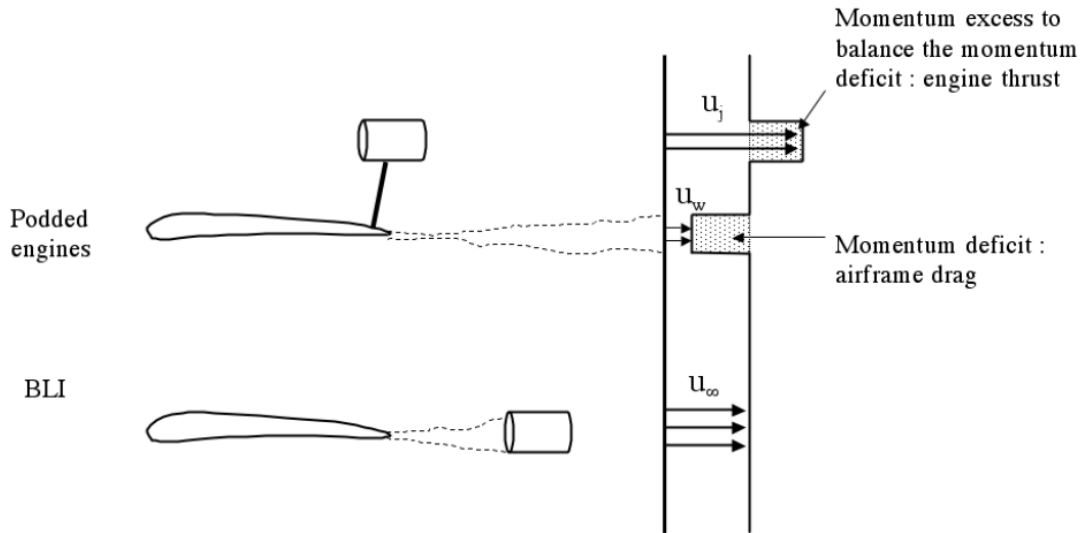


Figure 2-8 Podded and BLI Engine Configuration [71]

Plas et al assumed that the entire wake is ingested and the engine accelerates the wake back to freestream. Then, the following equation can be used to determine the force provided by the BLI engine:

$$F_{engine-BLI} = \dot{m} \cdot (V_j - V_w) = \dot{m} \cdot (V_0 - V_w) = D_A \quad (2-4)$$

Due to the decreased inlet velocity, the kinetic power P_{BLI} required by the BLI engine will be less:

$$P_{BLI} = 0.5 \cdot \dot{m} \cdot (V_j^2 - V_w^2) = 0.5 \cdot \dot{m} \cdot (V_0^2 - V_w^2) = F \cdot \frac{V_w + V_0}{2} \quad (2-5)$$

The difference in energy consumption between the two cases is due to the fact that for a given force, less power needs to be added to the flow entering the engine at a lower speed. Thus, the power required by the podded engine for a specific flight speed is higher than the power required by the BLI engine. The benefits of BLI can be described, then, as a reduction in the total power requirement of the vehicle due to an increase in propulsive efficiency.

2.3.1.2 BLI Inlet

Fundamentally, based on N+3 requirements, future concept aircraft using the BLI idea are predicted to have lower specific fuel consumption. By swallowing the boundary layer, it is feasible to lower engine specific consumption and overall aircraft drag. However, this is extremely dependent on the intake design and placement. The intake is extensively integrated into the airframe with the BWB form, which also helps reduce the aircraft's frontal size. Podded engines generate more drag as a result of the increased wetted area in the flow. Whereas the BLI concept is implemented by embedded intake that ingests a portion of the boundary layer over the fuselage's upper surface, as illustrated in Figure 2-9.

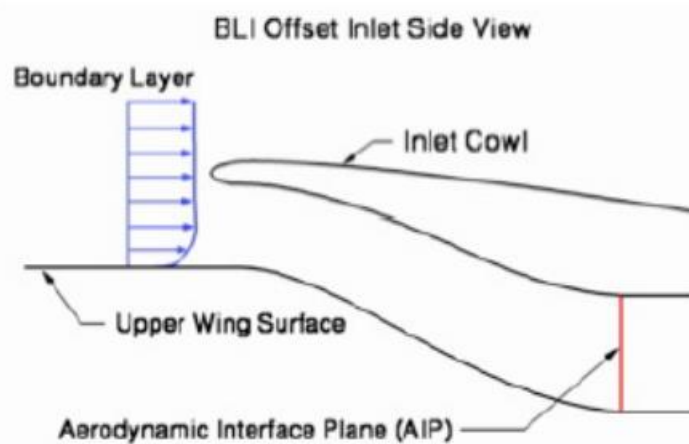


Figure 2-9 S-duct shape [72]

Embedded fans enable a greater BPR. However, BLI propulsion for BWB configurations adds performance to the inlet. As a result, it is recommended to

build the intake as an S-duct, which allows for improved inlet performance through optimisation. The S-duct inlet accepts both slow moving boundary layer flow and high velocity freestream flow. As illustrated in Figure 2-10, this leads in a non-uniform and complex flow [73]. A well-designed intake can effectively slow the approaching flow while retaining a high total pressure recovery and allowing for practically uniform flow. On the other hand, an increase in total pressure loss reduces propulsion efficiency, while an increase in flow distortion at AIP increases the fans' surge and stall limitations.

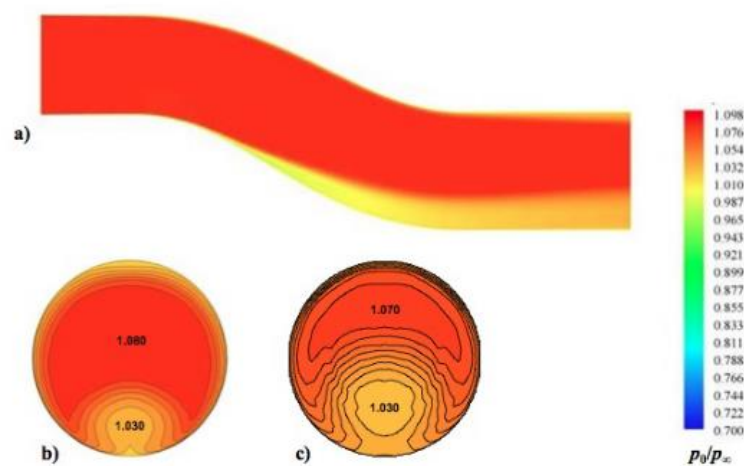


Figure 2-10 Flow Distortion in a S-duct [73]

2.3.1.2.1 Flow Distortion

Flow distortion is an important element to consider for the BLI concept for BWB configuration. With the application of BLI technology, it is necessary to measure the performance of the intake system using a quantified distortion parameter. Distortion associated to the inlet and engine performance can be classified into four categories:

- Total pressure distortion
- Swirl distortion
- Total temperature distortion
- Static pressure distortion

Due to the nature of the BWB intake, overall pressure distortion and pressure recovery are of key concern for S-ducts. A typical BLI S-duct contains circumferentially distributed zones of high and low total pressures. Circumferential distortion is defined by an angular shift in the total pressure throughout the intake [74]. The intake's upper half cross section contains high pressure levels indicative of undisturbed input flow, whilst the engine face's bottom section contains low pressure areas indicative of the ingestion of low momentum airframe boundary layer [75]. The flow distortion can result in a loss of fan efficiency and structural stability in the engine, which has an effect on the aircraft's performance. Avoiding flow distortion requires constructing the intake curvature at an appropriate design point. In general, this section will examine pressure recovery and fan face distortion as two critical elements to consider when analysing a duct for BLI. As these two parameters quantify the working conditions of the duct. Additionally, it can aid in optimising the duct design by altering the curvature. Increased Mach numbers, on the other hand, are deleterious to both inlet pressure recovery and engine face distortion [73][76]. The benefit of applying S-duct for embedded engines is that it requires less diffusion for the boundary layer, as flow slowdown occurs towards the fuselage surface due to the presence of viscosity. Static pressure gradients also influence the flow behaviour within a duct.

2.3.1.2.1.1 Total Pressure Distortion

Total pressure distortion occurs primarily as a result of wake and flow separation. Wakes are the disturbances caused by the interference of a fuselage boundary layer with intake shocks [74]. Frequently, when used for military purposes, the wake is ingested separately from the boundary layer. Likewise, flow separation can occur for a variety of reasons. Separation is facilitated in subsonic S-duct intakes principally by the aggressive centreline curvature [77]. This zone of separation is depicted in Figure 2-11. It originates as a result of centrifugal force between the boundary layer and the unfavourable pressure gradient in the bend [78].

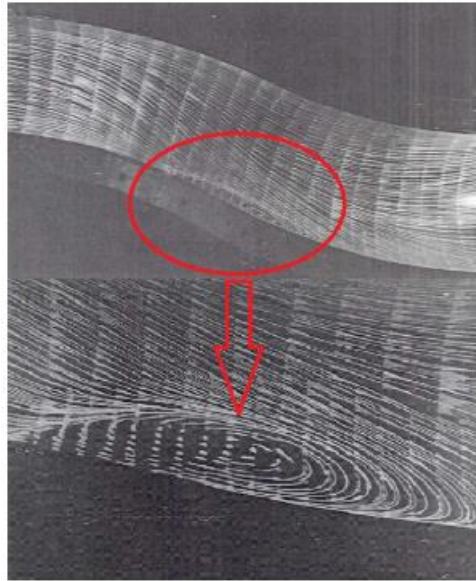


Figure 2-11 Flow Separation Region within S-duct [78]

Figure 2-12 illustrates the distribution of total pressure at the Aerodynamic Interface Plane (AIP) in a similar manner. At the lower section of AIP, the spoiled conical region with low total pressure can be clearly seen [17].



Figure 2-12 Total Pressure Distribution at AIP [78]

The distortion coefficient $DC60$. can be used to define total pressure distortion. $DC60$. quantifies the flow disparity by incorporating the distorted sector width specified by a critical-sector angle and the total pressure differential. As demonstrated in Eq. (2-6), $DC60$ can be defined as the average total pressure difference between the distortion zone of 60 and the AIP divided by the average dynamic head q_{AIP} at the AIP.

$$DC60 = \frac{P_{t,AIP} - P_{t,60}}{q_{AIP}} \quad (2-6)$$

where $P_{t,AIP}$ is the average total pressure at AIP and $P_{t,60}$ is the average total pressure at the distortion zone of 60°.

Rolls-Royce conducted research that resulted in the decision of the angle of 60 illustrated in Figure 2-13. This is because as the spoiled sector angle widens, the surge margin narrows. As a result, once the angle of 60 is reached, the surge margin reduction becomes invisible. As a result, this crucial angle of 60 is recommended as the minimum and is frequently used to determine engine performance [79]. $DC60$ has a maximum permissible value of 5% for embedded S-duct inlets in civil aircraft applications [80]. Previous research on the S-duct for the BWB distributed propulsion system demonstrated that it is practically impossible to decrease distortion to less than 5%. As a result, the permissible $DC60$ parameter value for the BWB distributed propulsion system is 10% [74].

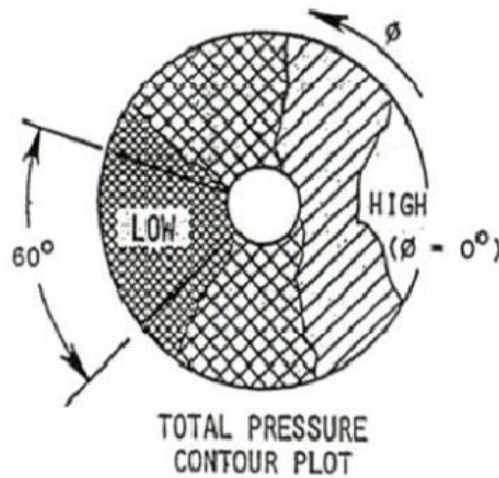


Figure 2-13 Total Pressure Distribution for a Given Plane [79]

2.3.1.2.1.2 Pressure Recovery

Pressure recovery is a critical attribute for a propulsion system since it aids in the evaluation of intake performance. Variation in the recovery of intake pressure has a direct effect on engine thrust. As a result, it is critical to analyse the pressure

recovery as well. Total pressure recovery is a measure of the RAM compression efficiency within the intake [74]. It is defined as the ratio of the average total pressure at AIP $\bar{P}_{t,AIP}$ to the reference total pressure $\bar{P}_{t,ref}$, which is commonly referred to as the mean total pressure at the intake face. The following equation is for pressure recovery:

$$PR = \frac{\bar{P}_{t,AIP}}{\bar{P}_{t,ref}} \quad (2-7)$$

Typically, the total pressure profile of the boundary layer is imposed at the intake, and the resulting mean value of total pressure at the inlet face is an integrated total pressure. According to previous study in this area, PR is greater than unity both within the duct and at the AIP [81]. As a result, an alternative approach can be employed for low-speed incompressible flow. This is based on the following equation.

$$PR = \frac{\bar{P}_{t,AIP} - \bar{P}_{t,ref}}{\bar{q}_{ref}} \quad (2-8)$$

Where:

$\bar{q}_{ref}$ is the freestream dynamic pressure.

Plas et al [4] offer a distortion evaluation for an S-duct and recognize that the most fundamental phenomena to represent are the inviscid behaviour of non-uniformities upstream and downstream of the fan. As illustrated in Figure 2-14, the more boundary layer is ingested, the greater the flow distortion. Additionally, the overall pressure applied to the fan face is decreasing as a result of the increased momentum deficit. The results of this study, acquired by high-fidelity CFD analysis and lower analytical calculations, indicate a 1-2% drop in fan efficiency in cruise mode for a given fan pressure ratio, due to a 4% reduction in the effective mass flow going through the engine duct. A further 1% decrease in compressor efficiency is caused by the flow blockage.

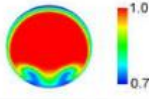
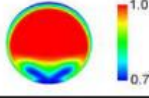
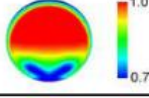
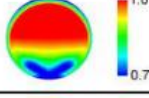
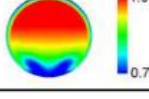
BLI	PR	DC(60)	Contours of P_t
10%	0.962	0.249	
20%	0.955	0.324	
33%	0.941	0.418	
45%	0.930	0.474	
59%	0.915	0.512	

Figure 2-14 Effect of BLI thickness on flow distortion [4]

2.3.1.2.2 S-ducts Design

The main objective of air breathing intakes is to deliver air at conditions suitable for the engine fan or compressor, supplied at the rate of mass flow required. The fan face conditions are often at a mismatch from freestream flow conditions, and the efficiency of this transformation process drives the design. An efficient intake recovers a large portion of the freestream total pressure at the engine interface by minimizing total pressure losses. Total pressure recovery is generally considered to be directly proportional to engine thrust in subsonic flow conditions [82].

Embedded engines allow for the inlet to remain flush with the airframe to take advantage of BLI while also providing some degree of noise suppression. Intakes for embedded engines often take the shape of S-ducts. The curvature in S-ducts diffuses the air faster than in conventional straight ducts, which leads to more compact designs and potential weight savings. With varying S-duct cross-sectional area and shape, the cross-sections can transition from rectangular, oval, semi-elliptical etc. at the entrance to the circular engine face. The general S-duct geometry and inlet shapes are shown in Figure 2-15. The centreline curvature of the duct can vary with different offsets and turning angles. S-duct

geometries are typically described by the offset and length ratios, based on the engine face diameter, and the diffusion ratio, which is the ratio of the outlet to inlet areas. An additional aspect ratio is used for rectangular inlets to define its width to height ratio.

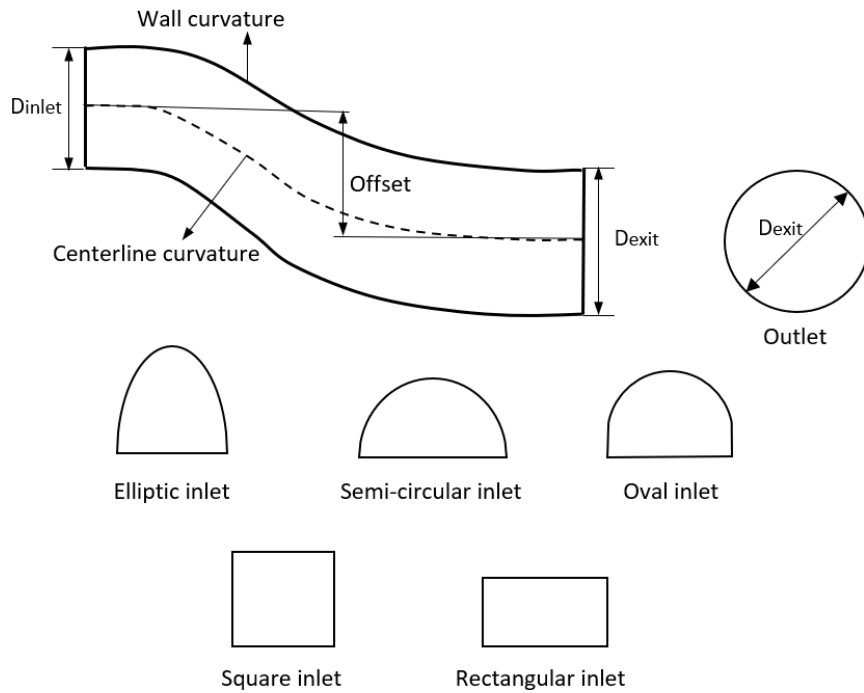


Figure 2-15 The general S-duct geometry and inlet shapes

Saha et al. [83] studied the effects of various cross-sectional shapes at the inlet of the intake duct. The shapes of interest were elliptical, ovular, semi-circular, square, and rectangular, and standard and renormalized group $k-\epsilon$ turbulence models were used. The elliptical-shaped inlet had the best performance, whereas the square-shaped inlet had the worst performance in terms of pressure recovery, loss coefficient, and flow distortion at the engine face.

Anand et al. [84] examined the effect of turning angle on the flow characteristics of S-shaped circular diffusers. The authors conducted experiments analysing three S-shaped circular diffusers with turning angles of $15^\circ/15^\circ$, $22.5^\circ/22.5^\circ$ and $30^\circ/30^\circ$ and found that the total pressure loss coefficient was independent of the turning angle.

Povinelli et al. [85] investigated the effect of diffuser centreline curvature and transitioning cross section, which is influenced by the change in geometry, which was semi-circular at the duct inlet and becomes circular at the engine face, on the use of the parabolized Navier-Stokes code. This analysis was conducted to determine the primary cause of flow distortion in the duct. The authors found that the centreline curvature was related to flow distortion and pressure loss but the transitioning cross section had a small impact on flow characteristics.

Gopaliya et al. [86] studied the performance characteristics of an S-duct by changing the inlet/outlet offset and the Reynolds number using computational analysis. Increasing the offset between the inlet and the outlet decreased pressure recovery.

The embedded engines with S-duct are affected by high levels of flow unsteadiness and distortion. This is driven by secondary flow and local flow separation due to the adverse pressure gradient typical of curved intakes. The main consequence of this characteristic is swirl and not uniform distribution of total pressure at the inlet of the compressor. This could potentially lead to unexpected stall and mechanical vibrations which can compromise the operational life of the entire propulsive system [87].

Multi-objective optimisation can be employed to deform the geometry of an S-duct and search for the best configuration which reduces flow unsteadiness and distortion.

Several research focus on the shape control that deforms the duct walls. A commonly selected S-duct for optimization is the Royal Aircraft Establishment M2129, introduced by researchers at British Aerospace Industries in the early 1980s [88]. It has offset, length and diffusion ratios of 0.9, 3 and 1.52, respectively. An S-duct optimization studied by Zhang et al. [89] featured an irregular bump on the upper wall after the first bend. The intent is to introduce a separated flow region far upstream that helps reduce cross-flow downstream of the duct. The two design parameters were the height and width of the surface perturbation. The objective function was the total pressure distortion at the engine interface. The inflow Mach number was 0.6, the Reynolds number based on the

inlet diameter was 2.6 million, and the BLI height was 7.3% of the inlet diameter. Using a gradient-based optimization algorithm, distortion was reduced by 63% compared to the baseline M2129 duct. Pressure recovery remained the same, as the bump behaved like a closed bubble and the separation was confined to that region.

With the same M2129 duct, D'Ambros et al. [90] and Chiereghin et al. [91] both used the Free Form Deformation (FFD) method [92] to control the S-duct geometry. Gradient-free genetic algorithms were used to search the design space. Considering pressure recovery and swirl as the two objectives, they were able to achieve considerable improvements over the baseline. Lee and Kim [93] used an adjoint method approach to optimize for two separate single objectives at a throat Mach number near 0.85 and a Reynolds number around 7.7 million based on the throat diameter. The first distortion minimization resulted in a 25% reduction from the baseline and the total pressure recovery increased by about 1%. The second optimization was a total pressure recovery maximization, which improved by 1.5% and decreased distortion by 3-7%.

Koo et al. [94] developed an aerodynamic shape optimization approach for a fixed rectangular BLI S-duct intake with boundary layer ingestion, intended for use on a high-subsonic, unmanned flight vehicle. The flight condition was a cruise Mach number of 0.7 and BLI that was 60% of the inlet height. The study was based on steady-state Reynolds-Averaged Navier-Stokes (RANS) simulations, and the geometry was controlled with an FFD volume. The goal of the optimization was to simultaneously maximize total pressure recovery and minimize distortion at the outlet, and they were able to achieve substantial improvements in both objectives. A convex Pareto curve was obtained showing the competing nature between total pressure recovery and distortion.

2.3.2 BLI Risks

Perhaps the most serious danger associated with BLI is the inlet and fan system's performance. While the intake system of an aviation engine is very simple, it is a critical component whose performance has a significant effect on the engine's specific fuel consumption. The intake must provide the needed amount of air to

the compression system without considerable losses or flow distortion, which could result in performance reduction or compression stability difficulties. Two risks exist as a result of this issue: 1.) Engine performance reduction (as a result of inlet total pressure recovery and fan distortion); 2.) Compression system instability owing to steady-state and dynamic inlet total pressure and swirl distortion. The efficiency of the intake is mainly determined by the distortion level of the intake air flow. When the intake boundary layer flow accounts for a large proportion of the total intake flow, which means that the distortion level of the intake air flow is high, such as high angle of attack flight, sideslip flight or in the presence of air turbulence, it will significantly reduce the total pressure recovery efficiency of the intake. When the undistorted flow further flows into the engine through the intake, the efficiency and pressure ratio of the fan will also be reduced, but a larger proportion of the boundary layer will help reduce the ram drag for the propulsion system, thereby improving the propulsion efficiency. When the intake boundary layer flow accounts for a small proportion of the total intake flow, such as during take-off, the engine needs to take in a large amount of airflow, and the mass flow of free flow is much larger than that of the boundary layer, which means the distortion level of the intake airflow is low, and the property of the intake mixed flow is basically the same as when the ingestion flow is completely free flow, so the benefit brought by the BLI are small. Thus, there is a risk that the gains in propulsive efficiency induced by the BLI will be cancelled out by the presence of a poorly performing inlet configuration, as well as a risk that the engine, while more efficient, will be incapable of operating within the required flight envelope for civil air transport. Nichols similarly highlights these hazards as the key sources of uncertainty for the concept in his study of the Silent Aircraft Initiative [95].

There are other dangers associated with the distortion, such as the effect on component degradation and longevity. These factors can have a major impact on operational expenses, perhaps cancelling out part of the fuel savings. Additionally, there is the challenge of developing a configuration with a highly integrated airframe and propulsion system. This worry is worsened by the fact that these components are manufactured by distinct firms, necessitating inter-

corporate collaboration in order to certify this new technology and ensure an economically viable design [96]. Additional details include the control of a system that is intentionally intended to exhibit steady-state and transient turbulence distortion during operation, as well as vibration caused by the same phenomenon. Significant research funding has been directed into developing a "distortion tolerant fan" capable of operating with increased efficiency and stall margin under the forms of distortion associated with BLI [97].

2.3.3 The Quantitative Benefits of BLI

Lynch's study in 1960 [98], one of the earliest on BLI, discovered 3% reduction in specific fuel consumption with application of BLI. In 2003, Dagget et al. conducted a research for the Boeing company as part of the project's Ultra-Efficient Engine Technology/Propulsion Airframe Integration. The purpose of this study was to determine the effect of BLI on BWB aircraft equipped with active flow control (AFC). The engine analysis was performed using the "ram drag" technique, which reduces the ram drag term contained within the net thrust by a percentage determined by the boundary layer characteristic averaged throughout its height. As a baseline engine, a podded engine based on conventional engine technology was built. To establish the fuel burn benefits of the ram drag reduction in the BLI design, a long S-Duct with a highly off-set embedded intake was used. It was determined that the baseline BLI configuration would improve fuel burn by 3.1%, which is a sizable improvement. The fuel burn advantage from the addition of the AFC technology was projected to be 5.5%.

Another research undertaken by Boeing examined the inlet configuration and the potential benefits associated with raising the inlet's aspect ratio. The approach and general configuration of the vehicle were very similar to those described in [99]. Additionally, the study refined the estimate of nacelle viscous drag by doing "proper analyses of viscous changes in which decreases in nacelle drag account for local Reynolds number effects." This accounting difference resulted in a significantly increased assessment of the potential for BLI with AFC and flush mounted inlets, which was previously estimated at 10%.

Plas et al [4] conducted research on boundary layer ingesting engines as part of NASA's silent aircraft initiative. This is the first study of its sort to investigate the effects of BLI using a real-world model of turbomachinery operating in non-uniform flow. The study made the assumption that the arrangement under consideration was a ducted fan configuration. The results indicated that using BLI on the aircraft under consideration for the silent aircraft initiative resulted in a 3-4% reduction in power consumption.

In 2009, Nickols and McCullers [100] conducted a configuration system analysis on the concept of the Hybrid Wing Body. This was a low-fidelity study that did not fully utilise engine design or cycle analysis methodologies. Rather than that, the study assumed a set percentage drag reduction on the aircraft's drag polar during the mission analysis, as well as a nacelle wetted area reduction factor, pylon weight structural component, and SFC penalty due to the reduced intake pressure recovery. The final analysis determined that the BLI technology resulted in a 5.2 percent reduction in fuel consumption.

Hardin et al. [101]undertook a system analysis of an aeroplane powered by a BLI arrangement. A detailed cycle model for an ultra-high-bypass propulsion system with BLI was included in the investigation. The system analysis used lower order models to simulate the many components of the BLI problem, including the theoretical benefits of BLI, the nacelle weight and drag, fan performance, and inlet pressure losses. Based on engine cycle calculations and weight estimates, aircraft trade factors were utilised to estimate fuel burn, and the study's findings indicated that a 3-5% BLI fuel burn benefit might be realised for the "N+2" generation aircraft when compared to a pylon mounted baseline. Another significant finding was that inlet pressure recovery and fan efficiency had a significant effect on the amount of fuel burned, with a 1% pressure recovery loss resulting in a 3% increase in fuel use. This demonstrates the need of low-loss inlets and distortion-tolerant fan topologies in maximising the technology's potential advantage.

Other ancillary benefits may exist, depending on the system design. For example, the weight of the propulsion system is often reduced due to the absence of an

engine pylon. There may also be a reduction in wetted drag, as the pylon and nacelle area can be reduced by properly embedded or flush mounted engines, which have a portion of their surface area contained within the aircraft and not exposed to the atmosphere, though this may be offset by aerodynamic interference or flow separation.

2.3.4 Modelling Approaches for BLI

This section will focus on describing the state-of-the-art system level approaches taken for each of the above components which require BLI impact modelling. It is necessary to note that the approaches described in this section are suitable for the BWB configuration with embedded propulsion and S-shape intake.

2.3.4.1 Airframe Aerodynamics and Boundary Layer

The literature review for the BLI theory turned out to be a function of the characteristics of the aerodynamics of the vehicle and specifically the boundary layer properties such as the momentum and kinetic energy thicknesses as well as the shape factors. This section will discuss the methods by which system studies have estimated the inviscid airframe properties as well as the boundary layer properties needed for the performance analysis. There are a few general methods used to generate the airframe boundary layers.

The two primary methods for conceptual level system studies are summarized in Table 2-1. Studies which use the first method include [102] [71], and studies which use the second type of method include [103] [101] [99]. The majority of system level studies avoid using CFD in the multi-disciplinary analysis loop, but rather assume that the boundary layers do not change much from the cruise point and simply use those profiles as a starting point. From the first method, typical approaches are to assume a Coles wake profile or a 1/7th power law profile which is typical of flat plate turbulent boundary layers.

Table 2-1 Comparison of two prominent methods for boundary layer characterization

Approach	Pros	Cons
1-D Boundary Layer Profiles	<ul style="list-style-type: none"> • Simple • Fast • Closed form solution • Scalable with Reynolds 	<ul style="list-style-type: none"> • Cannot capture 2-D/airframe effect • No angle of attack variation
CFD Based Boundary Layer Profiles	<ul style="list-style-type: none"> • Capture airframe effects 	<ul style="list-style-type: none"> • Not scalable • Data only at points where CFD is run • Expensive/time consuming

2.3.4.2 BLI Inlet Modelling

Figure 2-16 outlines the various regions of the flow field leading up to the fan face of the embedded engine. The pre-entry boundary layer region was described in the previous section along with the models that are typically employed for conceptual level studies. Additionally, there is the pre-compression region which is described by Plas in [104]. This is the region in which the streamtube is affected by the presence of the engine and the flow is compressed into the inlet capture area. Flow that is not ingested into the engine passes over the external cowl region which may typically induce some drag due to wall pressure or shock wave generation. Inside the inlet, the Mach number decreases as the flow is diffused inside the inlet until it reaches the fan face. The fan face Mach number is typically a function of the fan design specific flow capability.

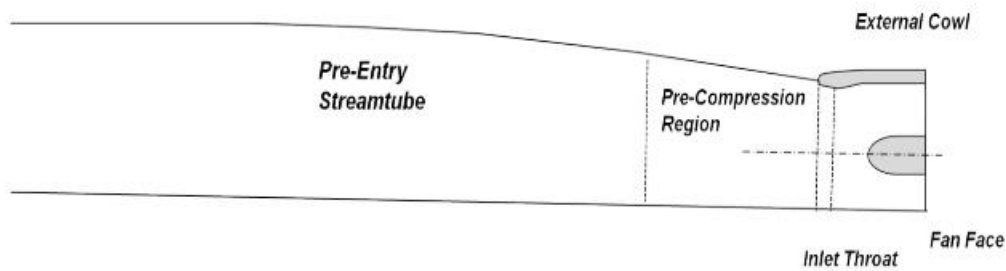


Figure 2-16 The regions of the flow field before the fan face [104]

Finally, the last component of the aerodynamic analysis prior to the fan face is the performance of the inlet duct. The necessary output of this analysis depends upon the fidelity of the fan model used. If the fan model requires more detailed fan face profiles, then a higher fidelity analysis must be used. If the fan model requires only a characterization of the dirty or distorted region, then a simple 1-D type analysis might suffice, such as the integral boundary layer method [104]. By far the most common approach, however, is the use of a simple inlet recovery parameter or inlet efficiency as is used in standard cycle analysis techniques. This means that the BLI losses are essentially passed as a lower mass averaged pressure recovery to the standard fan analysis. This approach is used in [103] [102] [101] [105] with various values assumed for the pressure loss, which tends to have a significant impact on BLI performance.

2.3.4.3 Fan Modelling

2.3.4.3.1 Empirical Method

At the conceptual, system study level, the fan modelling approach taken is typically a simple efficiency hit. This approach was used in [103] [101] [99][105] . Although simple, it does provide a basic parametric way to understand the impact of the fan performance relative to the BLI propulsive efficiency benefits to understand technology targets for a BLI fan design. Table 2-2 shows the differences in assumptions used for the fan efficiency for some of the important system studies mentioned earlier. It is common to assume that the efficiency penalty will be small, however the work conducted by Pratt and Whitney [97] shows that there is a likely efficiency penalty relative to a clean fan on the order of 0-1.5%.

Table 2-2 Fan efficiency assumption used for several system studies

Reference	η duct losses
Kawai, 2006 [99]	0%
Nickols, 2009 [105]	0%
Felder, 2011 [103]	1%
Hardin, 2012 [101]	0-8%

As discussed previously, Plas [71] conducted a study on 3 different levels of modelling fidelity for a ducted fan. For brevity, the details of each model will not be discussed here, but rather some of the key conclusions from the study will be summarized and some observations will be made that are relevant for the current work.

2.3.4.3.2 Parallel Compressor Method

The parallel compressor (PC) model is a standard “first approximation” for operability loss within the gas turbine industry when empirical or higher fidelity approaches are not possible or available [67] [16]. Recent studies have shown that modern gas turbine analysis tools can be integrated with a parallel compressor approach to yield reasonable and useful results for performance loss as well [52]. Furthermore, from an implementation stand-point, the only thing that is required is that the model needs to read two points from a compressor map, match the exit boundary conditions, and average the performance results.

Parallel Compressor Basics

The essential concept of the parallel compressor theory is to treat the singular distorted compressor as two uniform compressors each operating at their own level of inlet total pressure [57]. The low sector is the distorted region and operates at a lower total inlet pressure, while the high sector is simply the opposite and operates at the free-stream pressure. Each of the streams assumes uniform operation at an individual point on the “clean” or uniform compressor map. Figure 2-17 shows what happens in each of the sectors in terms of the overall compressor map. The standard assumption is that both sectors exhaust to a uniform static pressure plenum, which acts as the boundary condition that

the model must satisfy. Upstream of the compressor inlet, the presence of the compressor causes a redistribution of axial velocity and a change in static pressure, thus inducing static and total pressure distortion. The assumption that the exit static pressures are equal then requires that the low-pressure sector have a higher total pressure ratio than the high sector thus attenuating some of the pressure distortion.

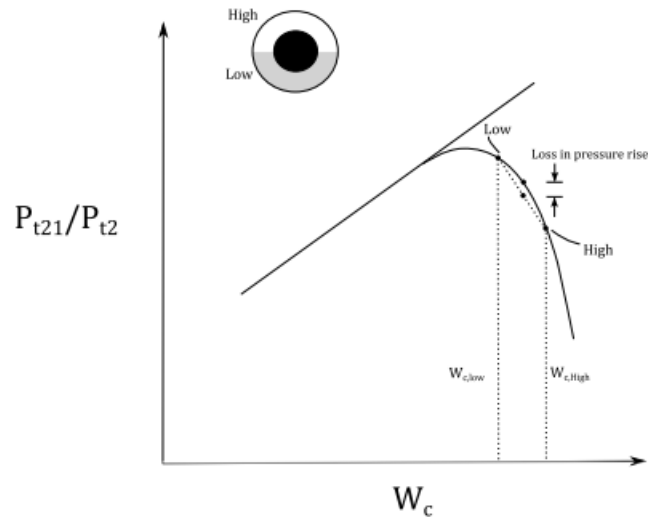


Figure 2-17 Diagram illustrating the parallel compressor concept [57]

The assumption of constant exit static pressure generally arises from the fact that the trailing edge flow angles after a compressor stage stator vane are roughly constant around the flow annulus, leading to a constant static pressure if the intermediate duct is straight and the flow is essentially 2D. Implicit in this assumption is that there is no circumferential flow redistribution between the sectors within the machine. For multi-stage compressors, this is reasonable if the circumferential length scales are large in comparison to a typical rotor-stator stage gap. For single stage fans, this assumption is valid since there is no gap within the compressor where flow can travel between the sectors. However, the assumption of constant exit angle is not typically valid, since the fan stator is some distance downstream of the splitter location in the bypass stream. Typical BLI fan high-fidelity analyses [29] show some amount of static pressure distortion at the fan exit which likely forces flow redistribution in the splitter gap and bypass duct. Furthermore, many studies have used the assumption for constant exit

static pressure when using the parallel compressor theory for fan analyses [69] [52] and the model used here will do the same assumption.

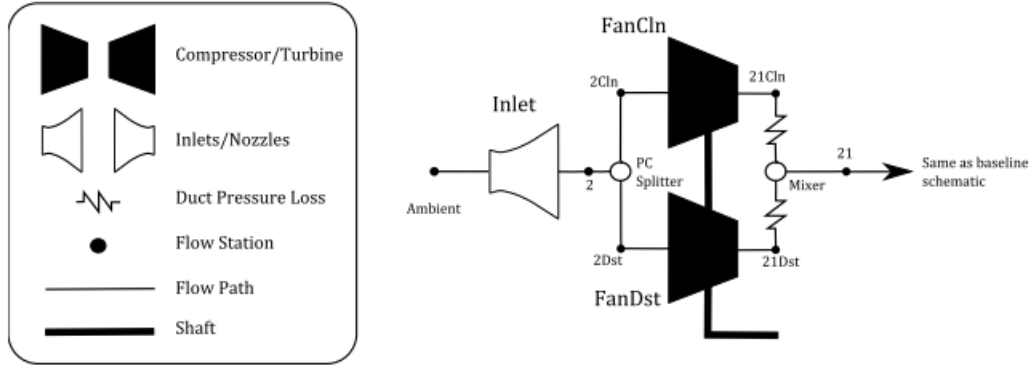


Figure 2-18 Parallel compressor engine model schematic

The basic parallel compressor model is described schematically in Figure 2-18. The model architecture consists of a splitter dividing the flow between the high and low sectors, one compressor for each sector, two ducts aft of the compressors which allow for pressure balance, and a mixer which mixes out the pressure and temperature between the sectors. The model aft of this station is precisely the same as the original engine model. The following sections will describe the essential assumptions made in dividing the flow and computing the operating lines of each compressor.

The first step in implementation is to split the flow and annulus area A_2 into the clean and distorted sectors. Assuming that a PC splitter is placed after the inlet, the mixed flow passing through the inlet is divided into clean and distorted sectors, and delivered to the corresponding compressors respectively. The distorted sector angle θ is an input to the PC splitter and the areas are split according to Eq. (2-9) and Eq. (2-10). Setting the sector angle θ to π (180 degrees) means the area will be split between the sectors. The area of clean sector A_{2Cln} is calculated by:

$$A_{2Cln} = \frac{A_2}{\left(1 + \frac{\theta/2\pi}{1 - \theta/2\pi}\right)} \quad (2-9)$$

The sum of the split areas has to equal the initial inlet out area. Thus, the area of distorted sector A_{2Dst} could be obtained by:

$$A_{2Dst} = A_2 - A_{2Cln} \quad (2-10)$$

The bypass ratio (BPR) for the PC splitter is defined as the ratio of the mass flow in the distorted and clean sectors, as given in Eq. (2-11).

$$BPR = \frac{\dot{m}_{2Dst}}{\dot{m}_{2Cln}} \quad (2-11)$$

The second step is to define the necessary static pressures for the different sectors at the inlet. This is done by assuming that the overall flow blockage (boundary layer displacement area) is kept constant but is all located within the dirty sector. The inlet exit discharge coefficient C_{d_2} is computed by the basic definition of C_d and from the inlet exit flow velocity:

$$C_{d_2} = \frac{\dot{m}_{actual}}{\dot{m}_{ideal}} = \frac{\dot{m}_2}{(\rho A u_e)_2} \quad (2-12)$$

Since there is, by definition, no blockage in the clean sector, the discharge coefficient is unity in that sector. All of the flow blockage is therefore contained in the distorted sector and the discharge coefficient $C_{d_{2Dst}}$ can be computed based on the sector mass flow definition.

$$C_{d_{2Dst}} = 1.0 - (1.0 - C_{d_2}) \frac{1.0 + BPR}{BPR} \quad (2-13)$$

The total pressure in the low sector is then set by making a similar assumption that the recovery in the high sector is unity and all of the recovery loss computed by the inlet model is contained in the low sector. The distorted sector loss ΔP_{t2Dst} is then computed as follows assuming mass flow averaged pressure.

$$\frac{\Delta P_{t2Dst}}{P_{t\infty}} = 1.0 - \frac{\frac{P_{t2}}{P_{t\infty}} (1.0 + BPR) - 1.0}{BPR} \quad (2-14)$$

The assumption is then made that the area split between the sectors is the same at the fan outlet as the inlet. The total exit area is adjusted to achieve an overall exit area ratio consistent with high-bypass fans.

$$A_{21Cln} = (A_{21Cln} + A_{21Dst}) \frac{A_{2Cln}}{A_{2Cln} + A_{2Dst}} \quad (2-15)$$

The mixer is used to mix out the losses between the sectors. The dirty sector is generally at some lower total pressure than the clean sector, implying there is some distortion transfer (pressure not fully recovered). The mixer requires that there be constant static pressure between the two mixer inlet stations, thus satisfying the constraint of the parallel compressor model. This is implemented in the solver, with the PC splitter bypass ratio used as the independent variable. This is to say that the corrected flow difference between the sectors is adjusted until the static pressure at mixer entrance is equal between the sectors.

Sector Map Treatment

The final challenge for the PC model is the treatment of the compressor maps between the two fan sectors. In the traditional PC model, the map must first be defined by a clean or uniform design. In this case, the general compressor performance map are used for both sectors. To define the scaled map for each design from the baseline map, an initial uniform pressure sizing case is run at the aerodynamic design point to set the map scalars. The map is defined by the clean sector, which is kept in “On-Design” mode during the clean ADP run, while the dirty sector always necessarily inherits the map scalars from the clean in every run. It is therefore kept in “Off-Design” mode regardless of the run type or design point. This process is outlined in Figure 2-19 below.

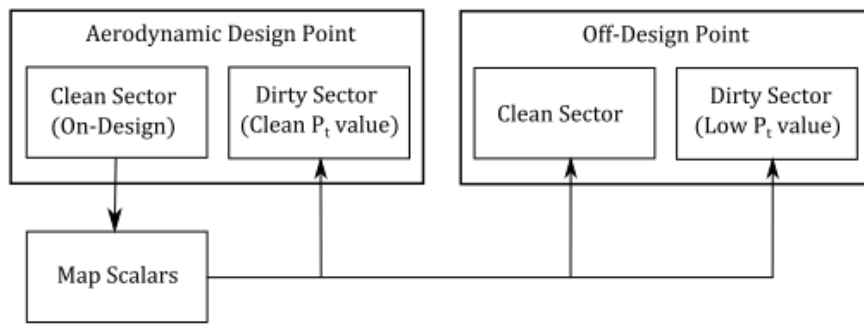


Figure 2-19 Parallel compressor map scalar production process

2.4 Identified Gaps in Knowledge

The review of the literature provided an overview of emerging developments in aircraft technology and the aviation business. For years, the commercial aviation industry's ongoing attempts to increase aeroplane performance have been mixed with an increasing desire to mitigate the industry's environmental impact. These objectives necessitate paradigm-shifting solutions, including novel aircraft configurations, novel technologies, and novel flight procedures. Among the several novel alternatives, two of the most promising are the Blended Wing Body (BWB) design and boundary layer ingest technology. Through the preceding review of the literature, two critical gaps in study have been discovered that this research seeks to solve.

Multidisciplinary design and optimisation techniques have been widely used in the aircraft conceptual design process. Qin et al. [106] carried a three-dimensional aerodynamic surface optimisation of the BWB with three-dimensional aerodynamic surface optimisation of the BWB. The results show an improvement in the aerodynamic performance for the given BWB platform and design cruise condition. Antoine and Kroo [65] used a Multi-disciplinary Optimisation (MDO) framework to investigate the viability of decreasing a commercial airline's operating costs while meeting noise and pollution limit. However, the above research did not consider the packaging problem when optimizing the shape of the BWB aircraft. The literature review on BWB revealed that the strongly linked configuration of the BWB introduces a packaging issue that, if not addressed early in the design phase, may become a larger one later

in the process. To overcome these obstacles and examine the effect of integrating advanced systems and concepts, it will be necessary to construct a multivariate design synthesis and optimisation tool for the conceptual design of the BWB that incorporates a robust packaging procedure. This tool will be used to explore for the design space of the BWB configuration to provides ideas and references for the subsequent design of the BWB aircraft.

NASA developed a hybrid wing body configuration named SUGAR Ray for NASA's N+3 goals [7]. In the comparison with the conventional tube-and-wing aircraft, different performance simulation methods were selected, for example CASES (Computer Aided Sizing and Evaluation System) database, which is a heritage empirical Douglas application, was used to develop the aerodynamic buildups for conventional aircraft and SUGAR Ray reference drag buildups uses wing-body CFL3D RANS. Different simulation methods will bring errors to the results. The comparison for different types of aircraft in the GENUS design environment could eliminate errors caused by method selection, and thus can better quantify the advantages of BWB aircraft and BLI technology.

During the course of the research, a further gap of BLI in literature was identified. Previous research in literature sets the propulsion system as an independent research objective to explore the potential improvement of engine performance by BLI technology. Therefore, the emphasis of these studies is on design point sizing and performance. Several research has taken the propulsion system as the object and conduct simulations on the design point, which shows that fuel savings in the region of 5-10% can be achieved by using boundary layer ingestion [101][107]. However, The working state of the engines changes with the changes in the flight conditions of the aircraft. To estimate the benefits of BLI technology more accurately, the aircraft and the engine should be studied as a whole. As research on aircraft with BLI propulsion progresses, it becomes necessary to develop a BLI propulsion system modelling workflow and incorporate it into the aircraft design process.

3 BWB Design Methodologies

This section presents the detailed methodologies selected, developed, and integrated for the design, analysis and optimisation of blended wing body aircraft.

3.1 The GENUS Aircraft Conceptual Design Environment

3.1.1 Introduction

This section introduces the MDAO aircraft design environment, named GENUS [109], which is the main design tool used and developed throughout this research. The GENUS framework has been developed by the Aircraft Design Group at Cranfield University since 2012. It is a comprehensive computational design tool that enables the reuse of design processes and integrates multiple disciplinary models in a modular, scalable, flexible, and independent arrangement. Its name comes from the taxonomic categorization of biological entities (as illustrated in Figure 3-1), and it refers to the capability of modelling and analysing various aircraft species within a common framework.

To date, GENUS has been successfully applied to the conceptual design and analysis of conventional airliners [110], hypersonic vehicles [111], solar unmanned aerial vehicles, supersonic business jets [112] and combat unmanned aerial vehicles [113].

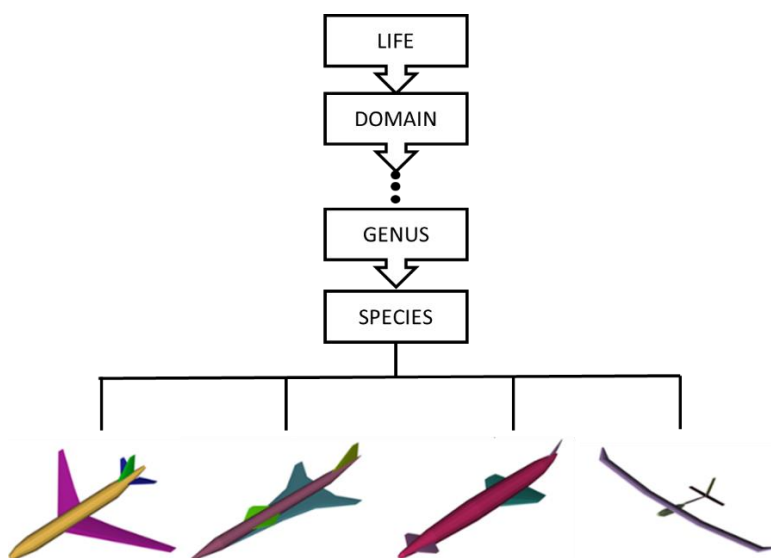


Figure 3-1 GENUS geometry representation of various aircraft species

3.1.2 The GENUS Framework

GENUS is a Java-based architecture that is composed of nine critical modules, as shown in Figure 3-2: 1) geometry, 2) mission, 3) propulsion specification, 4) mass breakdown, 5) aerodynamics, 6) propulsion, 7) packaging and C.G., 8) performance, and 9) stability. Additional modules can be added to satisfy the requirements of particular aircraft designs. Additionally, this framework for design space exploration incorporates gradient-based and custom-made genetic algorithms.

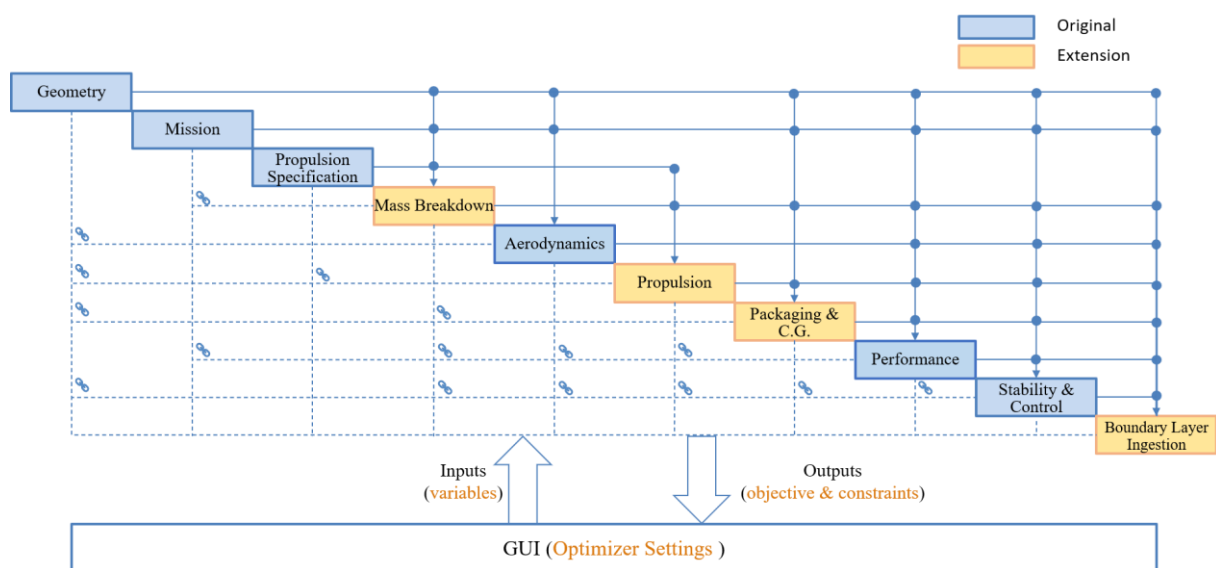


Figure 3-2 Modules and data flow in GENUS

The modules and optimiser are integrated into a GUI, which is developed by the JAVA native Swing GUI. The GUI enables a knowledgeable user to select the modules, set the inputs, define internal variables, objective function and constraints.

In the Module Selection tab, as shown in Figure 3-3, the designer can select different submodule/methods for the integrated study. As shown in Figure 3-4, the Inputs tab gets inputs from the designer. These inputs can also be pre-set in the codes to reduce repeatability. There are three types of inputs: list type (such as engine type, fuel type, etc.), value type (such as wingspan, fuselage length, etc.) and int value type (such as wing kinks, number of engines, etc.).

In order to generate the numerical outputs from each analysis module, the user must perform a Single Run, which produces a single instance of the aircraft based on the current inputs. The result of each module and the plot of current design can be checked in the single run and plot tab, as shown in Figure 3-6.

As for the optimisation setting, the inputs checked in the Optimise column will be variables for the optimisation process, whose lower boundary and upper boundary can be set. The objective function and the appropriate constraints are then selected in the Outputs tab, as shown in Figure 3-5. The Optimise tab is shown in Figure 3-7, which is used to select whether the objective function is to minimised or maximised, set the types of constrains and choose the optimisation algorithm. A graphical output of the objective function's history is shown in the same tap.

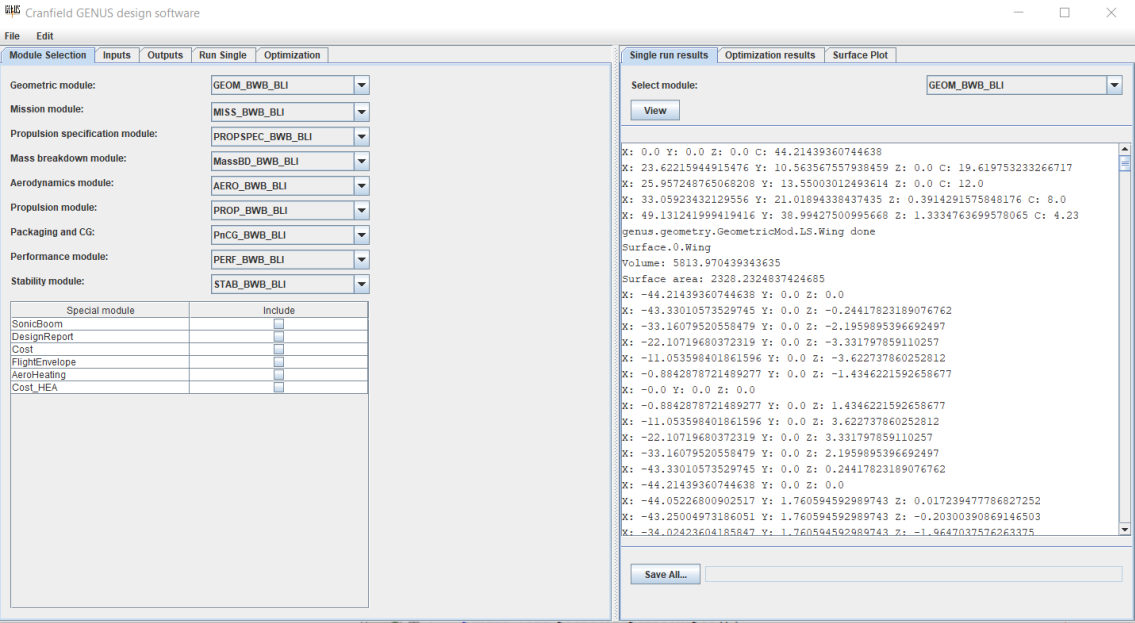


Figure 3-3 GENUS GUI for Module selection

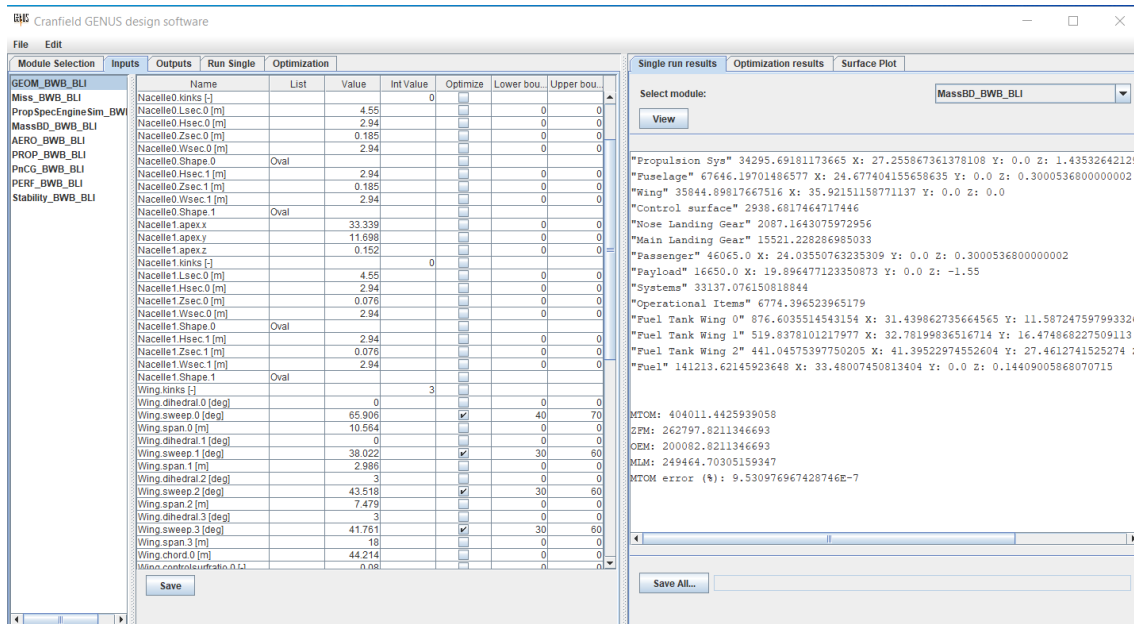


Figure 3-4 GENUS GUI inputs tab

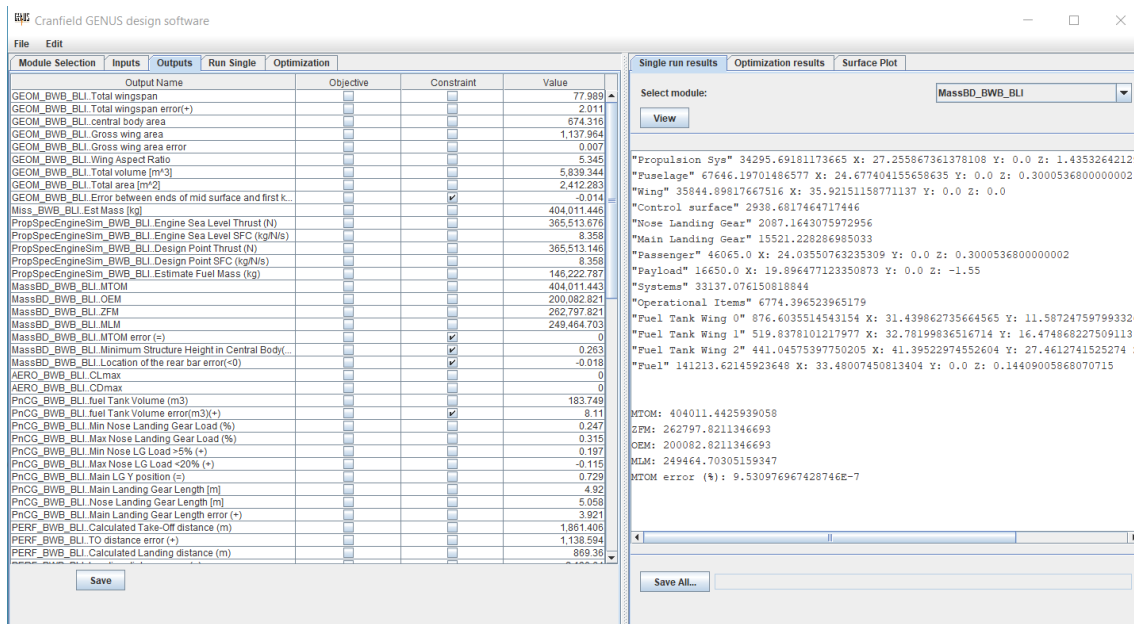


Figure 3-5 GENUS GUI outputs tab with objective function and constraint selected

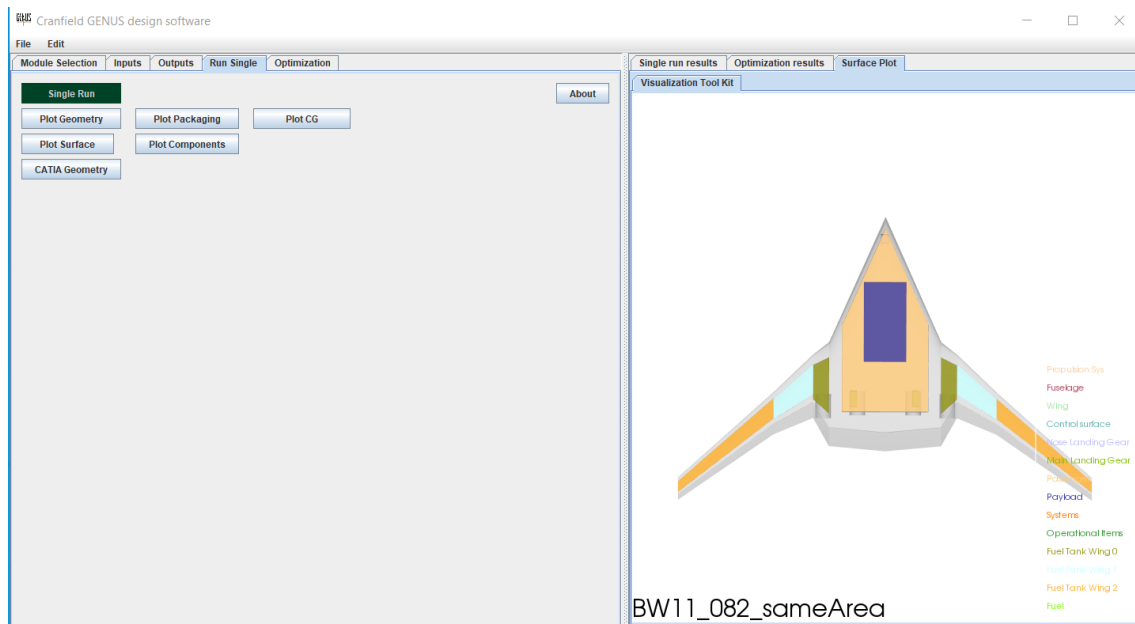


Figure 3-6 GENUS GUI single run and plot tab

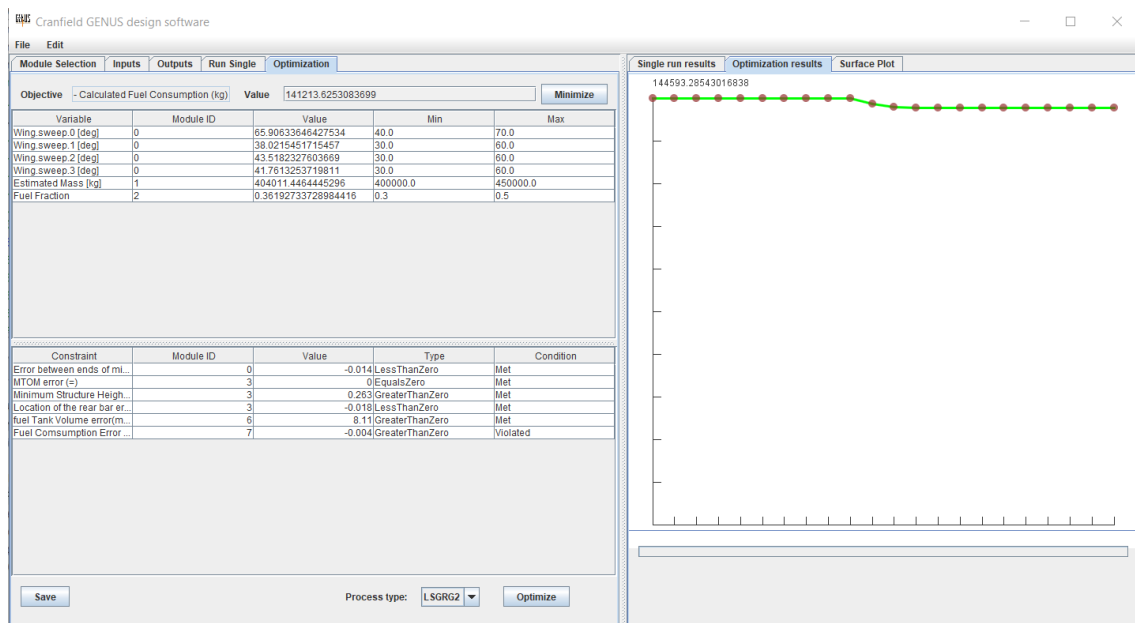


Figure 3-7 GENUS GUI optimisation output

3.2 Geometry

Effective and efficient parameterization or general description of aircraft parameters is critical for overall aircraft performance analysis and design space exploration research, especially when considering the potential complexity of

aircraft-level technology and performance evaluation, as well as data exchange between multidisciplinary optimisation disciplines.

Aircraft geometries in the GENUS framework are specified through discrete elements which are divided into body components and lifting surface. Each class can be further into sub-classes which use a library of cross section shapes, platform dimensions, twists, dihedrals, and aerofoils for a full geometry definition. The input parameters to define a BWB geometry are listed in Table 3-1.

Table 3-1 Wing-body input parameters

Input Variables	Variable Type	Description
Preset geometry	List value	Choose a preset geometry in the list
Lifting Surface number	Integer value	Total number of lifting surfaces
Wing kinks	Integer value	Number of kinks of lifting surface
Wing chord	Real value	Chord length of the lifting surface section
Wing aerofoil type	List value	Aerofoil type of lifting surface
Wing aerofoil parameters	Real value	aerofoil parameters of lifting surface section
Wing span	Real value	Span of lifting surface section
Wing dihedral	Real value	Dihedral of lifting surface section
Wing incidence	Real value	Incidence of lifting surface section
Wing sweep	Real value	Sweep of lifting surface section
Wing control surface ratio	Real value	Control surface ratio of lifting surface section

Through the definition of the above parameters, the geometry can be generated by VTK (Visualization Toolkit) [114], which is an open-source software for 3D computer graphics. The wire frame and surface geometry plot of a BWB airplane is shown in Figure 3-8. GENUS can also generate a list of points that can be directly used to visualize in CATIA.

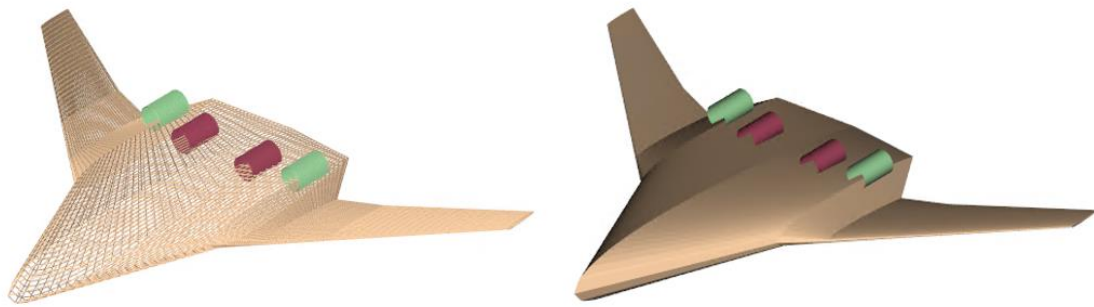


Figure 3-8 Wire frame and surface geometry plot

Furthermore, a variety of geometry formats required for different analysis tools can be used and transformed from these discrete components. Examples of such formats are the XYZ point-cloud, the LAWGS format, DATCOM geometry, AVL panels and PANAIR structured grids.

3.3 Mission

The mission module acts primarily as a user input module where some of the basic parameters required for the analysis of the design mission are specified. Designers can set their own mission variables in the subclass module. Table 3-2 lists mission input for a typical BWB commercial airliner which is the same as Airbus A-380 class ‘Very Long Rang’ standards.

Table 3-2 Mission input parameters

Input	Value	Units
Cruise Altitude	10059	m
Cruise Speed (TAS)	254	m/s
Cruise Mach	0.85	-
Target Range	14000	km
Number of Passengers	555	-
Number of Crew	6	-
Estimated Take-off Mass	478256	kg
Manoeuvre Load Factor	2.5	-
Take-off Distance	3000	m
Landing Distance	2500	m

3.4 Mass breakdown Estimation

Mass prediction is a critical step in the aircraft design process. It has an impact on both the cost and performance analysis of an aircraft. Generally, statistical data or physics-based estimations methods are selected to estimate the component weight during the conceptual aircraft design stage. GENUS has implemented several Class II or statistical methods including Howe's mass prediction method[115], Raymer's mass prediction method [20], Torenbeek's method [23], and Cranfield in-house mass prediction method, etc.

It should be noted that the majority of approaches are only valid for a given range or aircraft. For unconventional aircraft, such as the BWB, necessary modifications should be made based on the basis of the original methods to increase the accuracy of estimates. One important difference between BWB aircraft and conventional tube-and-wing configurations is the pressurised cabin modelling. A sizing method proposed by NASA [116] is applied in this research. Additionally, due to the scarcity of available data, it is not possible to determine the weight of the BWB cabin (centre body) and aft centre body directly from statistical regression. Bradley chose a surrogate approach based on Finite Element Analysis to conduct the modelling process.

The mass estimation equations for systems, structure components, and miscellaneous item are given in Appendix B.

3.5 Propulsion

3.5.1 Propulsion Specification

The propulsion specification module describes the engine type and fuel to be used, as well as the propulsion system requirements, which provides the flexibility to combine various types of propulsion systems needed for complex vehicles. Figure 3-9 shows the structure of the propulsion specification module in the GENUS with the main subclass named *Powerplant*. Each Powerplant subclass represents one type of propulsion system which can be modelled independently under different requirements.

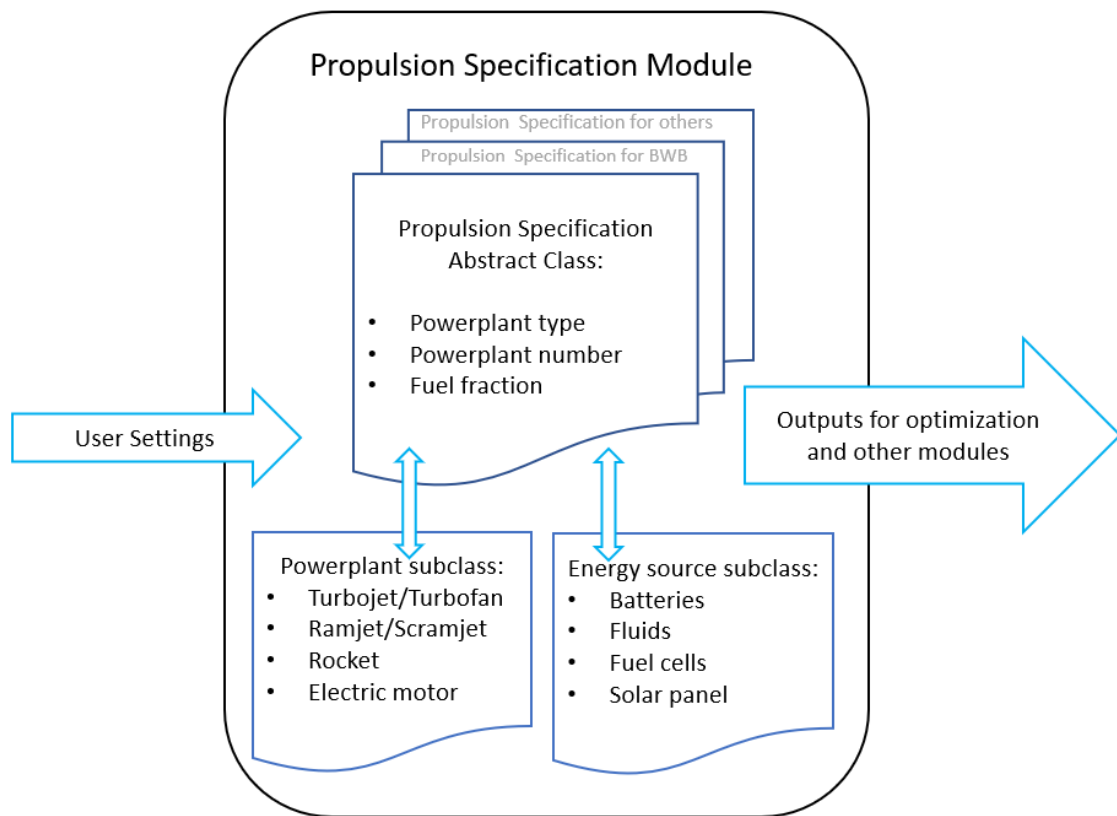


Figure 3-9 Propulsion specification within GENUS

3.5.2 Propulsion modelling

According to the input of various propulsion system parameters in the propulsion specification module, the propulsion module calculates the combined propulsion system performance at any flight condition. Figure 3-10 shows the hierarchy and data flow of this module. Two matrices are used to store the available thrust and the matching specific fuel consumption (SFC). By interpolating the Thrust-Mach-Altitude-Throttle matrix and the SFC-Mach-Altitude-Throttle matrix, users can obtain the propulsion system performance for a given flying state.

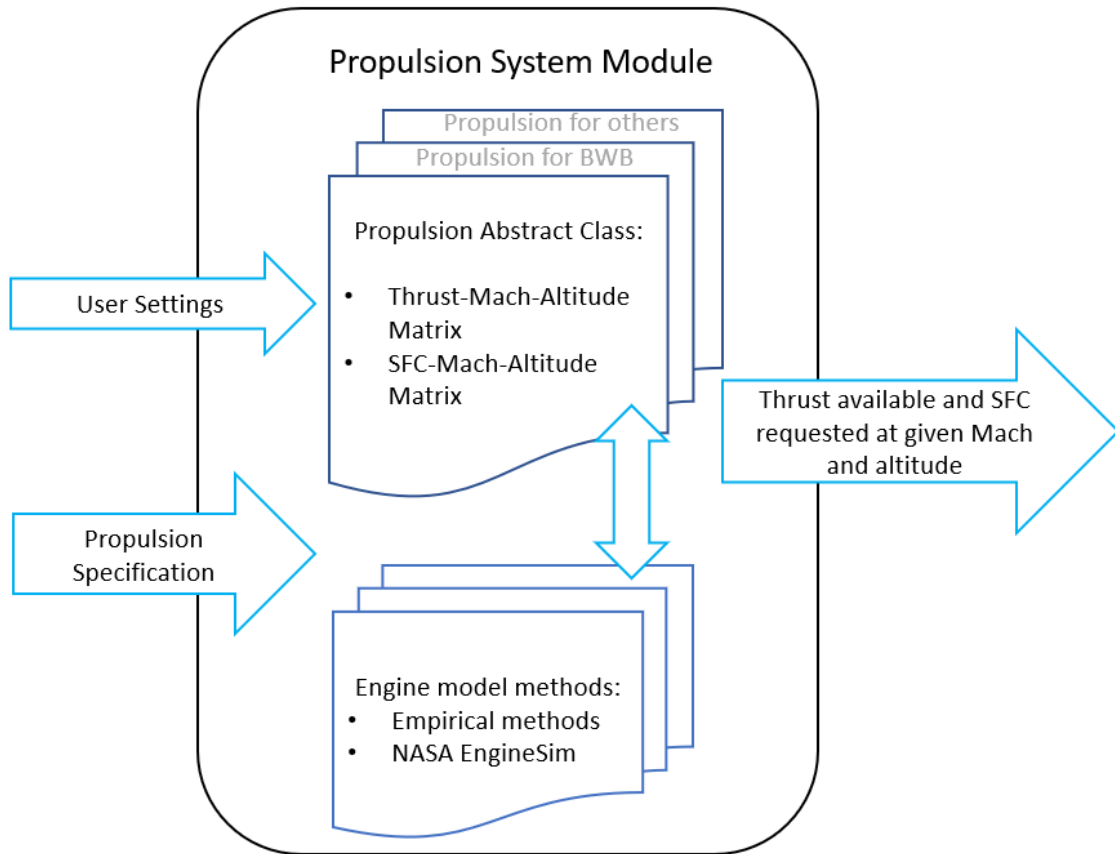


Figure 3-10 Hierarchy and data flow in propulsion System module

In terms of the method selection, there are some low-fidelity propulsion simulation methods from textbooks [20] [23] [115], where thrust and fuel consumption are a function of a few engine characteristics such static sea-level thrust, bypass ratio and flight conditions. They are efficient but low accuracy. In this research, several engine analysis tools has also been reviewed in Table 3-3 and it was conducted that the NASA EngineSim, whereas being the simplest of the tools mentioned below, has sufficient capability while also maintaining the open source, license free philosophy of GENUS. Thus, the EngineSim has been adapted in GENUS to conduct the engine performance calculation. It estimates the engine performance based on engine parameters, including the geometry size, pressure ratio and efficiency of each component, material properties, flight condition, throttle setting and so on, which provide more freedom for engine design and the possibility for the integration of boundary layer ingestion model.

Table 3-3 Engine modelling tools

Tool	Open Source	Capabilities
EngineSim	✓	Educational/Gas turbine analysis
Gasturb	×	Gas turbine performance analysis
NPSS	×	Component design and optimisation
TurboMatch	×	Gas turbine performance analysis
pyCycle	✓	Thermodynamic cycle modelling library for OpenMDAO framework

EngineSim works in two modes, design mode and test model. In the design mode user could defines the design point for the engine and then evaluates the off-design performance in its test mode. The EngineSim generic engine design procedure has shown in Figure 3-11.

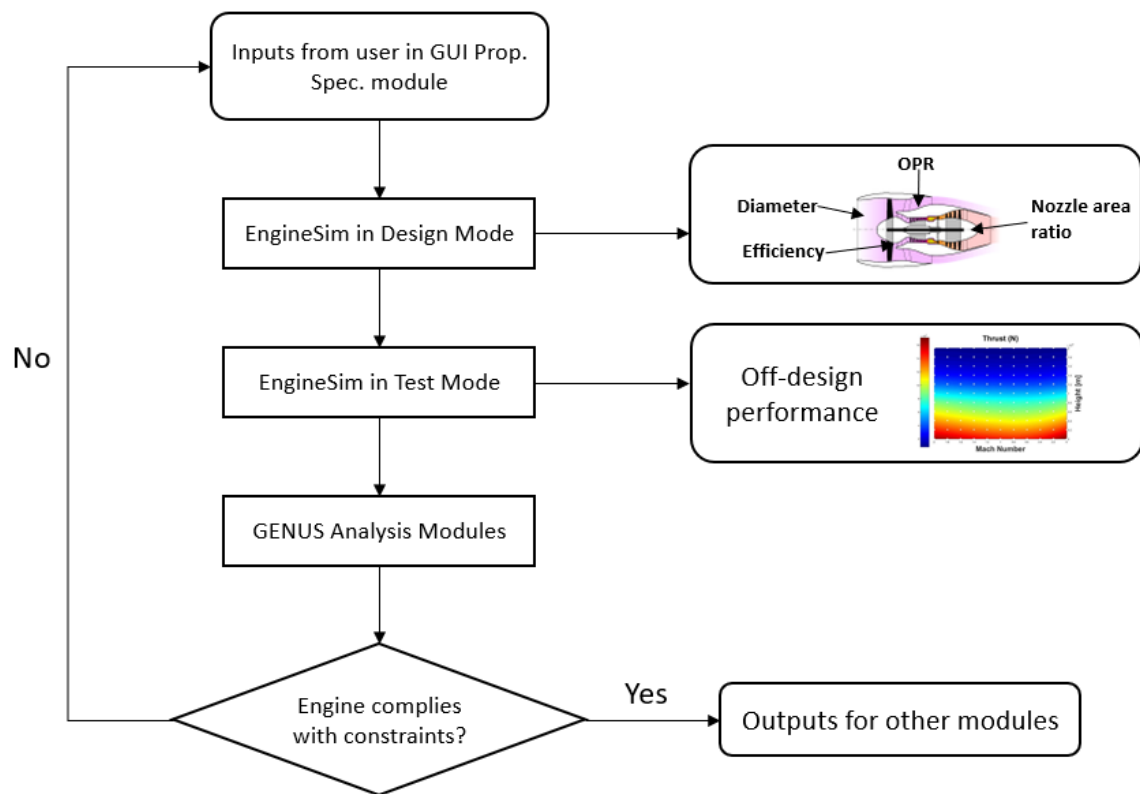


Figure 3-11 EngineSim generic engine design procedure

Take the turbofan engine as an example, the cycle analysis for turbofan engines is given in Eq. (3-1) to (3-16) based on the two-spool turbofan engine model shown in Figure 3-12. and the required inputs are listed in Table 3-4.

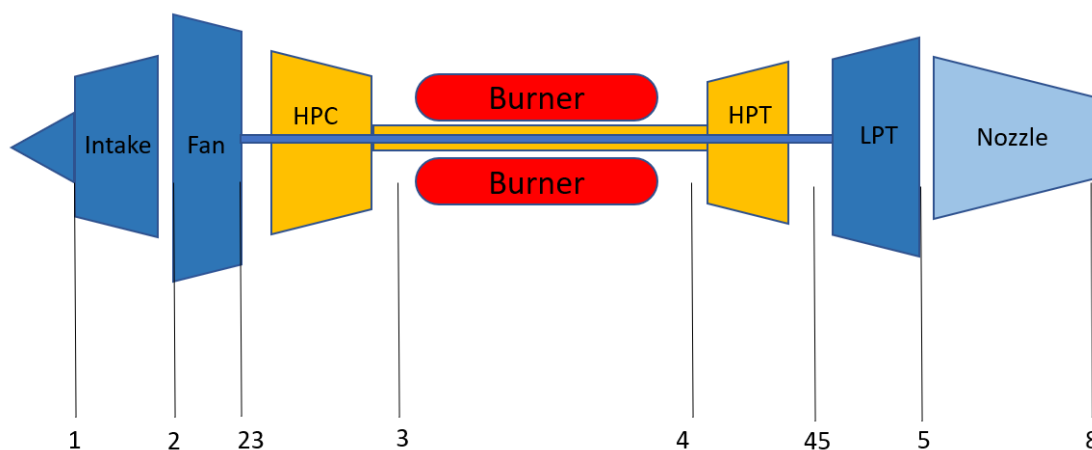


Figure 3-12 Two spool turbofan engine configuration

Table 3-4 Input parameters for propulsion model

Input Variables	Description
H	Altitude
Ma_0	Flight Mach number
BPR	Bypass ratio
FPR	Fan pressure ratio
CPR	Compressor pressure ratio
η_c	Compressor efficiency
TET	Turbine exit temperature
η_b	Bruner efficiency
TPR	Turbine pressure ratio
η_t	Turbine efficiency
η_n	Nozzle efficiency

Starting with the calculation to the free stream at a given altitude and Mach Number by using an International Standard Atmosphere (ISA), which is listed in Appendix A. Then total temperature and pressure at inlet could be calculated from Eq. (3-1) and (3-2).

$$\frac{T_{01}}{T_a} = 1 + \frac{\gamma - 1}{2} Ma^2 \quad (3-1)$$

$$\frac{P_{01}}{P_a} = \left(\frac{T_{01}}{T_a} \right)^{\frac{\gamma}{\gamma-1}} \quad (3-2)$$

Where γ is the specific heat ratio, T_a and P_a are the ambient static temperature and pressure respectively. As intake does not do any work, the total temperature from inlet to the fan remains constant and the pressure recovery coefficient could be calculated from the USAF standard shown in Eq. (3-3).

$$\frac{P_{02}}{P_{01}} = \begin{cases} 1.0 & Ma < 1.0 \\ 1 - 0.075 * (Ma - 1)^{1.35} & 1.0 < Ma < 5.0 \\ \frac{800}{Ma^4 + 935} & Ma > 5.0 \end{cases} \quad (3-3)$$

Pressure and temperature across the compressor are calculated as:

$$CPR = \frac{P_{03}}{P_{02}} \quad (3-4)$$

$$\frac{T_{03}}{T_{02}} = \frac{1}{\eta_c} \left[\left(\frac{P_{03}}{P_{02}} \right)^{\frac{\gamma-1}{\gamma}} - 1 \right] \quad (3-5)$$

The pressure in the burner remains nearly constant during burning, decreasing by only 1 to 2 percent. The value of TET is set by the throttle with the maximum temperature determined by material limits.

$$\text{Burner pressure ratio} = \frac{P_{04}}{P_{03}} \quad (3-6)$$

$$T_{04} = TET \quad (3-7)$$

Then the pressure and temperature across the turbine are calculated as:

$$\frac{T_{05}}{T_{04}} = 1 - \left(\frac{T_{03}}{T_{02}} - 1 \right) \frac{T_{01}}{T_{04}} \quad (3-8)$$

$$\frac{P_{05}}{P_{04}} = \left[1 - \frac{1}{\eta_t} \left(1 - \frac{T_{05}}{T_{04}} \right) \right]^{\frac{\gamma}{\gamma-1}} \quad (3-9)$$

The engine temperature ratio (ETR) and engine pressure ratio (EPR) can be calculated as:

$$ETR = \frac{T_{05}}{T_{04}} \frac{T_{04}}{T_{03}} \frac{T_{03}}{T_{02}} \quad (3-10)$$

$$EPR = \frac{P_{05}}{P_{04}} \frac{P_{04}}{P_{03}} \frac{P_{03}}{P_{02}} \quad (3-11)$$

The exit pressure at the nozzle is assumed to be the ambient pressure, and the nozzle flow is assumed as choked at the minimum area A_g . The exit velocity is then given by:

$$v_e = \sqrt{2C_p T_{05} \eta_n \left[1 - \left(\frac{P_a}{P_{05}} \right)^{\frac{\gamma-1}{\gamma}} \right]} \quad (3-12)$$

Where:

C_p is the specific heat coefficient.

The mass flow at the core nozzle can be obtained by the choked nozzle condition as:

$$\frac{m_{dot} \sqrt{\theta_t}}{\delta_t A_8} = \sqrt{\gamma} \frac{P_0}{\sqrt{T_0}} M \left[1 + \frac{\gamma-1}{2} M^2 \right]^{\frac{-(\gamma+1)}{2(\gamma-1)}} \quad (3-13)$$

The specific net thrust can be obtained from the freestream and exit velocities as:

$$\text{specific thrust} = \frac{F_n}{m_{dot}(1 + BPR)} \quad (3-14)$$

Where:

θ_t is the ratio of the total temperature to the reference total temperature.

δ_t is the ratio of the total pressure to the reference total pressure.

The fuel to air ratio can be determined by:

$$f = \frac{\left(\frac{T_{04}}{T_{03}} \right) - 1}{\left(\frac{Q_R}{C_p T_{03}} \right) - \left(\frac{T_{04}}{T_{03}} \right)} \quad (3-15)$$

Where:

Q_R is the fuel heating value.

Finally, the thrust specific fuel consumption is given by:

$$TSFC = \frac{f m_{dot}}{F_n} \quad (3-16)$$

Figure 3-13 shows the validation of EngineSim fully integrated into the GENUS framework for a low-bypass turbofan. The reference data [117] is from Jack Mattingly's theory. The results indicate that the engine performance from GENUS are close to the results of reference data and the mean absolute percentage error (MAPE) of thrust and TSFC between these two methods is 18.25% and 8.48% respectively, which helps to validate the codes.

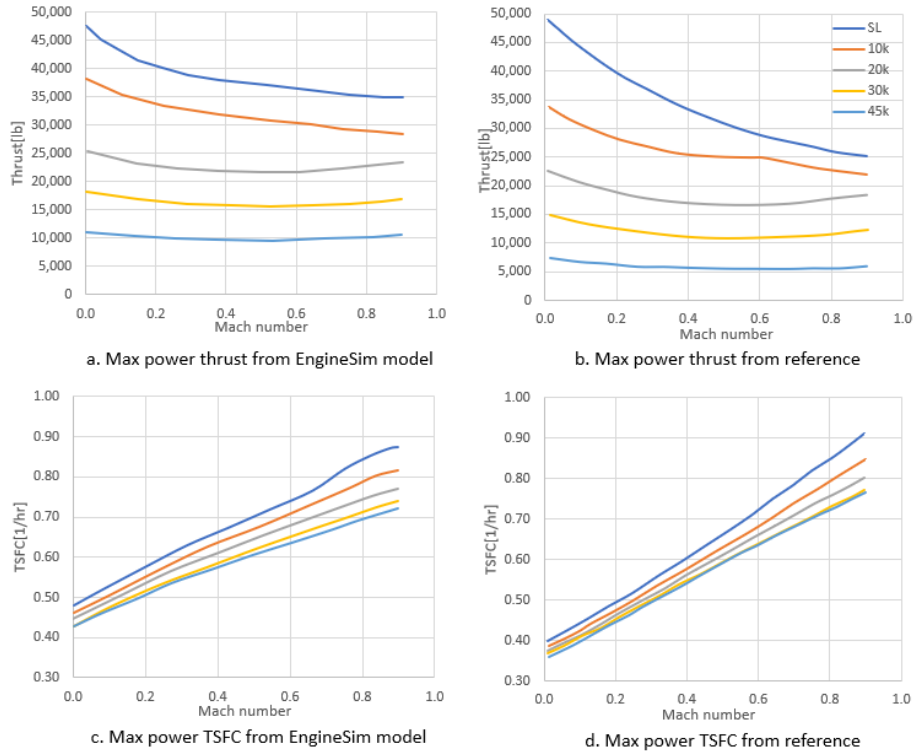


Figure 3-13 Comparison of low-bypass turbofan uninstalled engine performances

3.6 Aerodynamics

Aerodynamic analysis is a critical component to the aircraft conceptual design process. The GENUS aerodynamics module predicts the aerodynamic characterisation of a flight vehicle through the combination of several analysis methods and fidelity levels. The results are stored in a coefficient matrix as the data objects shared by other modules.

The total drag coefficient C_{Dtotal} is classified as the induced drag C_{Di} , the friction drag C_{Df} , the wave drag C_{Dw} and the trim drag C_{Dtrim} , as exemplified in Eq. (3-17). Among them, the trim drag is estimated in the performance module.

$$C_{Dtotal} = C_{Df} + C_{Di} + C_{Dw} + C_{Dtrim} \quad (3-17)$$

3.6.1 Friction Drag

Skin friction and form drag calculations have been adapted from a component-based approach proposed by Gur et al. [118] due to its parametric definition based on the geometric characteristics, the flexibility to include laminar to turbulent transitions and the flexibility to include form factors from various sources.

The total friction drag is obtained by Eq. (3-18). And the effects of flow turbulence are estimated by Schlichting's composition formula [119] in this research with the laminar transition fraction F_{trans} shown in Eq. (3-19). For laminar flow, the Eckert reference temperature method [120] is used, and for turbulent flow, the Van Driest II method [121] (based on the von Kármán–Schoenherr model) is used. The complete rigorous model for the flat-plate skin-friction coefficient can be found in [122].

$$C_{Df} = \sum_{j=1}^N \frac{FF_j C_{Fj} S_{wetj}}{S_{ref}} \quad (3-18)$$

$$C_{Fj} = C_{Fturb} - F_{trans} [C_{Fturb} - C_{Flam}] \quad (3-19)$$

Form factors are given in Eq. (3-20) and (3-21) for body components and lifting surfaces, from Torenbeek [23], chosen as conservative factors compared to other sources such as Hoemer and Raymer [20].

$$FF_{BodyComp} = 1 + 1.5 \left(\frac{d}{l} \right)^{1.5} + 50 \left(\frac{d}{l} \right)^3 \quad (3-20)$$

$$FF_{LiftSurf} = 1 + 2.7 \left(\frac{t}{c} \right) + 100 \left(\frac{t}{c} \right)^4 \quad (3-21)$$

3.6.2 Wave Drag

Compressibility effects must be accounted for during high subsonic and transonic flight segments, especially when operating beyond the drag divergence Mach number (Ma_{dd}) which is defined as the Mach number that results in a drag increase of 20 drag counts ($C_{Dw} = 0.002$).

The Ma_{dd} is obtained using the Korn equation extended with simple sweep theory as shown in Eq. (3-22).

$$K_{Aerofoil} = Ma_{dd} \cos \Lambda_{0.5} + \frac{C_l(y)}{10 \cos^2 \Lambda_{0.5}} + \frac{t/c(y)}{\cos \Lambda_{0.5}} \quad (3-22)$$

where $\Lambda_{0.5}$ is the midchord sweep angle and $K_{Aerofoil}$ is the Korn factor ranging from 0.95 for supercritical aerofoils to 0.87 for regular aerofoils. The validation of Eq. (3-22) is given in [123] for various thickness ratios and airfoil types. Figure 3-14 shows the effect of aerofoil types. For $Ma = 0.85$ and 10% thickness ratio, the increased maximum drag divergence lift coefficient due to improved aerodynamics is large, and perhaps more interestingly, the optimum sweep angle is reduced for the advanced aerofoil compared with the conventional aerofoil. Figure 3-15 shows the large effect of thickness ratio. It shows clearly that the thin wings has a higher maximum drag divergence lift coefficient. In addition, the use of thicker wings means that the optimum sweep is higher than the thin wing, but not dramatically.

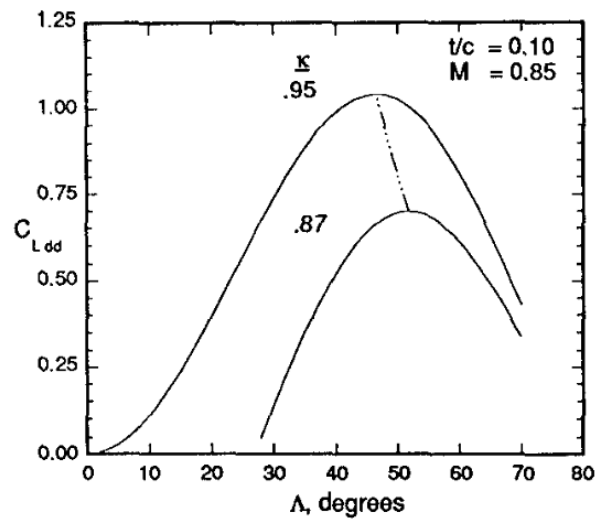


Figure 3-14 Effect of Korn factor on maximum drag divergence lift coefficient [123]

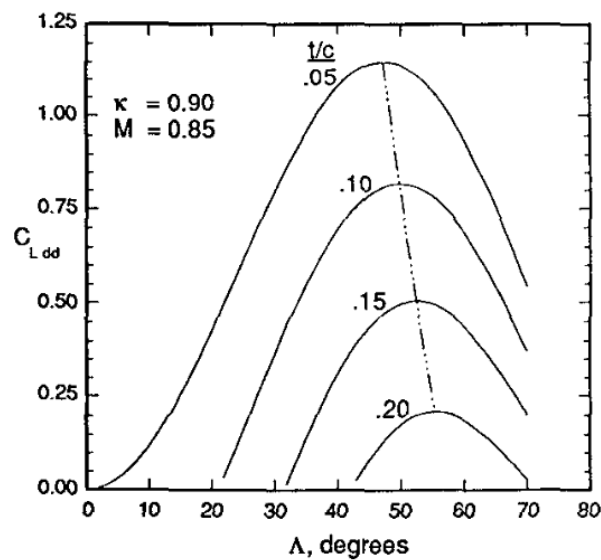


Figure 3-15 Effect of aerofoil thickness on maximum drag divergence lift coefficient [123]

According to [118], the local critical Mach number is lower than the drag divergence Mach number by a constant 0.108 as shown in Eq.(3-23).

$$Ma_{crit} = Ma_{dd} - 0.108 \quad (3-23)$$

The drag rise can be modelled based on the Lock's fourth power law [124] as the following function of the Mach number:

$$C_{Dw_{strip}} = \begin{cases} 0 & Ma \leq Ma_{crit} \\ 20(Ma_{\infty} - Ma_{crit}(y))^4 & Ma > Ma_{crit} \end{cases} \quad (3-24)$$

The contribution of the wave drag on a wing strip relative to the total three-dimensional wave drag is calculated using the area ratio S_{strip}/S_{ref} :

$$C_{Dw} = \sum_{j=1}^n C_{Dw_{strip_j}} \frac{S_{strip_j}}{S_{ref}} \quad (3-25)$$

The present wave-drag calculation procedure starts by dividing the wing into strips. Each strip is represented by its two-dimensional thickness ratio t/c , lift coefficient C_l and half-chord sweep $\Lambda_{0.5}$. Then the drag divergence Mach number is calculated using Eq. (3-22). Next, the local critical Mach number is found using Eq. (3-23), and the wave-drag coefficient on each strip $C_{Dw_{strip}}$ is found using Eq. (3-24). Finally, the three-dimensional wave drag is calculated using Eq. (3-25).

3.6.3 Induced Drag

Several aerodynamic analysis tools including digital DATCOM [48], an advanced vortex lattice method AVL [125], panel method PANAIR [126] and Hypersonic Arbitrary Body Program SHABP [127] has been integrated into the GENUS framework, which allow users to select a reasonable method to perform aerodynamic analysis on a particular aircraft.

Due to the unique geometry of the BWB aircraft, digital DATCOM is not suitable to perform the main aerodynamic analysis because it relies on equivalent wing planforms and its method consists of a compendium of data mapped through simple semi-empirical relations. Similarly, SHABP is mainly used for hypersonic analysis and AVL is limited to subsonic speeds. The main advantage of GENUS is to better enable comparisons between conventional and advanced concepts and configurations to be made including solar UAVs, supersonic business jets and combat UAVs flying in different speed regime. Therefore, the panel method PANAIR, which enable compute the aerodynamic characteristics of different

configuration flying in subsonic to supersonic regime, was selected as the aerodynamic analysis tool for BWB configuration.

PANAIR was originally developed by Boeing and NASA, which can perform aerodynamic analysis of arbitrary three-dimensional geometries by solving the Prandtl-Glauert equation shown in Eq. (3-26)

$$(1 - M_{\infty}^2)\phi_{xx} + \phi_{yy} + \phi_{zz} = 0 \quad (3-26)$$

In order to derive Eq. (3-26) from the Navier-Stokes equations, the following flow assumptions have to be made:

- The viscosity terms have been eliminated
- The flow is irrotational
- Steady state conditions
- Small perturbation approximation

A comprehensive explanation of potential flow theory and its implementation on the PANAIR code is out of the scope of this research, for more detail refer to the user manual [126].

3.6.4 Surrogate Model

Through the combination of the aforementioned aerodynamic analysis tools, a full aerodynamic characterisation is available at any given flight condition. The aerodynamic data can be interpolated between angle of attack, Mach number, and type of coefficient stored in the Coefficients Matrices, as schematically shown by the surrogate contours and experimental points in Figure 3-16 Aerodynamics surrogate model and experimental points.

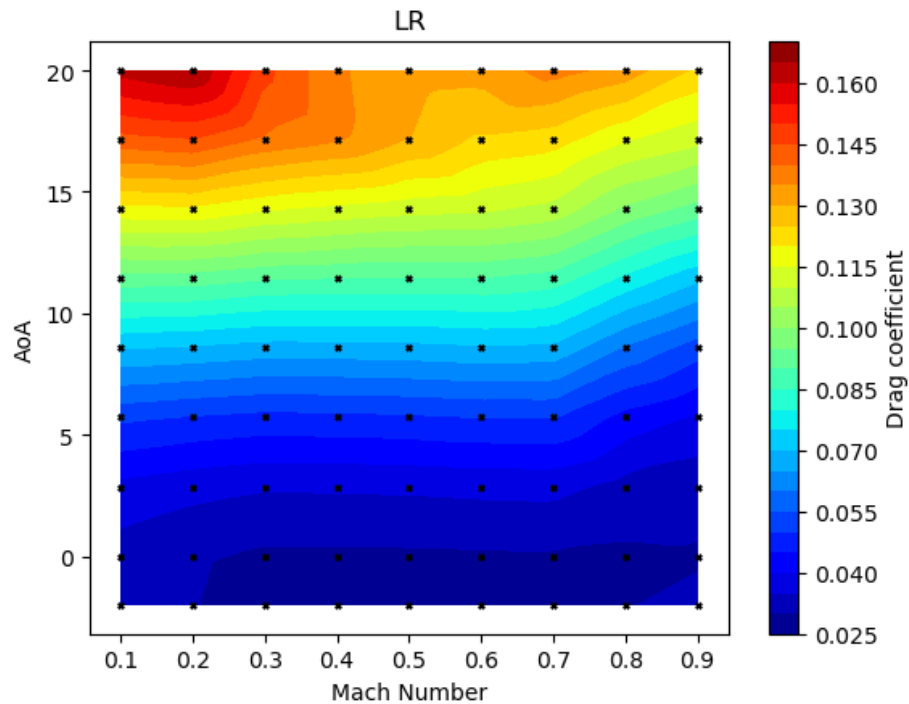


Figure 3-16 Aerodynamics surrogate model and experimental points

Furthermore, visualisation of pressure coefficients is available through an external post-processing package called pyNastran, as shown in Figure 3-17 pyNastran post-processor result for PANAIR.

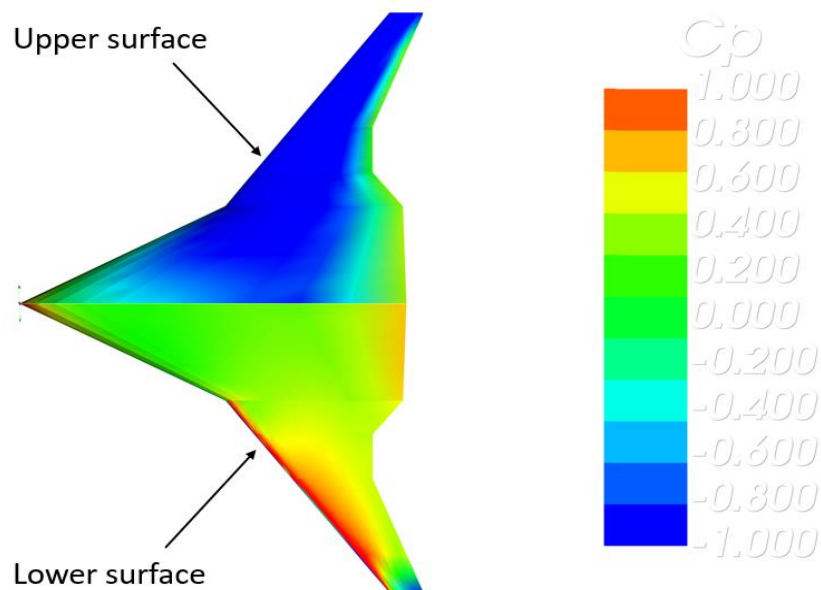


Figure 3-17 pyNastran post-processor result for PANAIR

3.7 Performance

The field performance is evaluated in the performance module, which also allow users to add additional mission constrains, such as manoeuvrability and point performance requirements for the following optimisation work.

The complete flight mission is discretized into sufficiently tiny mission parts, which are then analysed using Newton's equations of motion, with the aircraft acting as a mass point. Power equilibrium must be maintained, which means that the net power of thrust and drag must be converted to potential and kinetic energy changes. Within the mission calculations, an iterative method is used to account for variable changes throughout the mission segments. The aerodynamic coefficients and engine performance characteristics are interpolated for the duration of the flight mission, which is based on the flight Mach number and altitude.

Mission segments have been incorporated into the performance module to create more realistic operating conditions. The lift-to-drag ratio, in particular, has been analysed as a function of real airspeed, flight altitude, and the current aircraft mass, which is updated by the preceding mission segment's fuel consumption. Likewise, specific fuel consumption changes according to flight speed, altitude, and engine thrust, which is determined by the current aerodynamic drag.

Each mission segment is broken into dozens of smaller sections, with the aircraft travelling at a uniform acceleration in each segment. The entire mission is iterated upon until the design range is attained. As operating conditions also have a significant impact on aircraft fuel consumption. To prevent unnecessary operational implications, fuel consumption has been estimated using predetermined flight profiles that are typical for each aircraft type. Flight restrictions such as flight physics limitations and engine performance constraints are satisfied by the predefined flight profile.

3.8 Packaging and Centre of Gravity

Packaging and CG module is a novel and essential section in the GENUS environment. Its functions are to calculate the location of the centre of gravity of

the various mass components, and mass configurations of the vehicle (take-off mass, operational mass, zero fuel mass, etc.).and to enable all internal objects fitting in the surface.

The CG module calculate the location of the centre of gravity of the various components, and mass configurations of the vehicle for various conditions (with and without payload and fuel) through the Eq. (3-27) to (3-29). This functionality is highly dependent on the definition of the mass breakdown components preciously set in the Mass breakdown module. A series of parameters define each mass component, including the mass, name, centre of gravity location, a volume, a shape, and the dimensions. Subsequently, the results of this module is requested by the Stability and Control module for the longitudinal stability analysis.

$$X_{CG} = \frac{\sum m_i x_{cg_i}}{\sum m_i} \quad (3-27)$$

$$Y_{CG} = \frac{\sum m_i y_{cg_i}}{\sum m_i} \quad (3-28)$$

$$Z_{CG} = \frac{\sum m_i z_{cg_i}}{\sum m_i} \quad (3-29)$$

Packaging is a critical aspect of BWB design due to the irregular cross-section of the configuration. It ensures there is sufficient internal space to accommodate all components within the geometry's constraints and avoids clashes between the components and outer shell of the vehicle. The packaging layout of the typical BWB is shown in Figure 3-18 The packaging layout of the typical BWB.

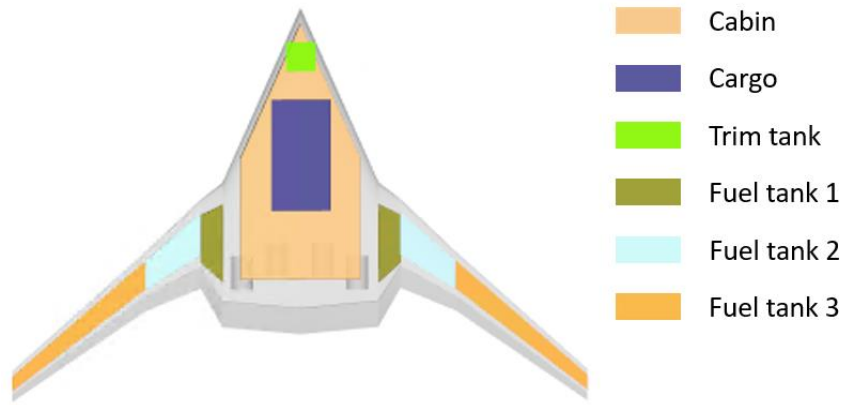


Figure 3-18 The packaging layout of the typical BWB

Clashes can be detected and resolved by calculating the distance between every defined component and the outer surface. This action is performed through a series of helper methods which compare the corner coordinates of the known component against the coordinates of the outer shell, which is shown in Figure 3-19. The minimum structure height in Eq. (3-30), meaning the lowest value of these distance, can be defined as a more than zero inequality constraint all the inner components to fit in the outer shell.

$$h_{struc_{min}} = \min\{h_1, h_2, h_3, h_4 \dots\} > 0 \quad (3-30)$$

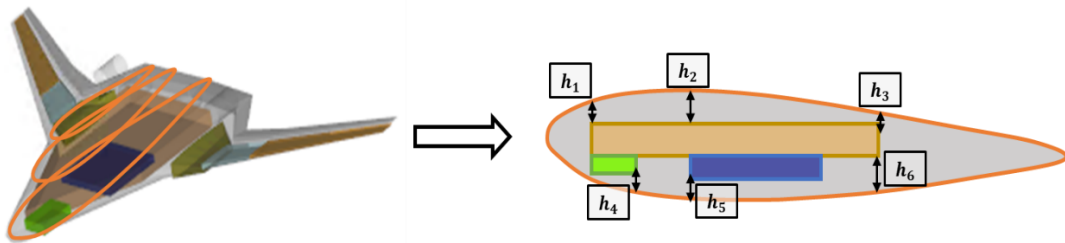


Figure 3-19 Overview of airfoil fitting to the cabin geometry

The flow chart in Figure 3-20 exemplifies the general execution procedure of the packaging module within a multidisciplinary design and analysis loop. In this research, the components are represented by simple three-dimensional shapes with rectangular cross sections, which are not consistent with the actual shape of the components. The simple rectangular shapes increase the structure height at

the corner point and increase the thickness of the aerofoils to accommodate all the components. The increase in aerofoil thickness can lead to excessive internal redundant volume and adversely affect aerodynamic performance. Therefore, in practice, the internal components of the BWB are designed to be curved, and the curved shape can avoid the high non-linear stress problems caused by the box shape. Taking the cabin as an example, several popular structural concepts are all designed to be curved at the edge area, such as the separate pressure shell and the integrated skin and shell concepts. However, specific structural design problems should be solved in the process of detailed design. Conceptual design stage mainly provides the distribution of internal components and preliminary shape design. Simple three-dimensional shapes can be used to represent these internal components for the rapid conceptual simulations.

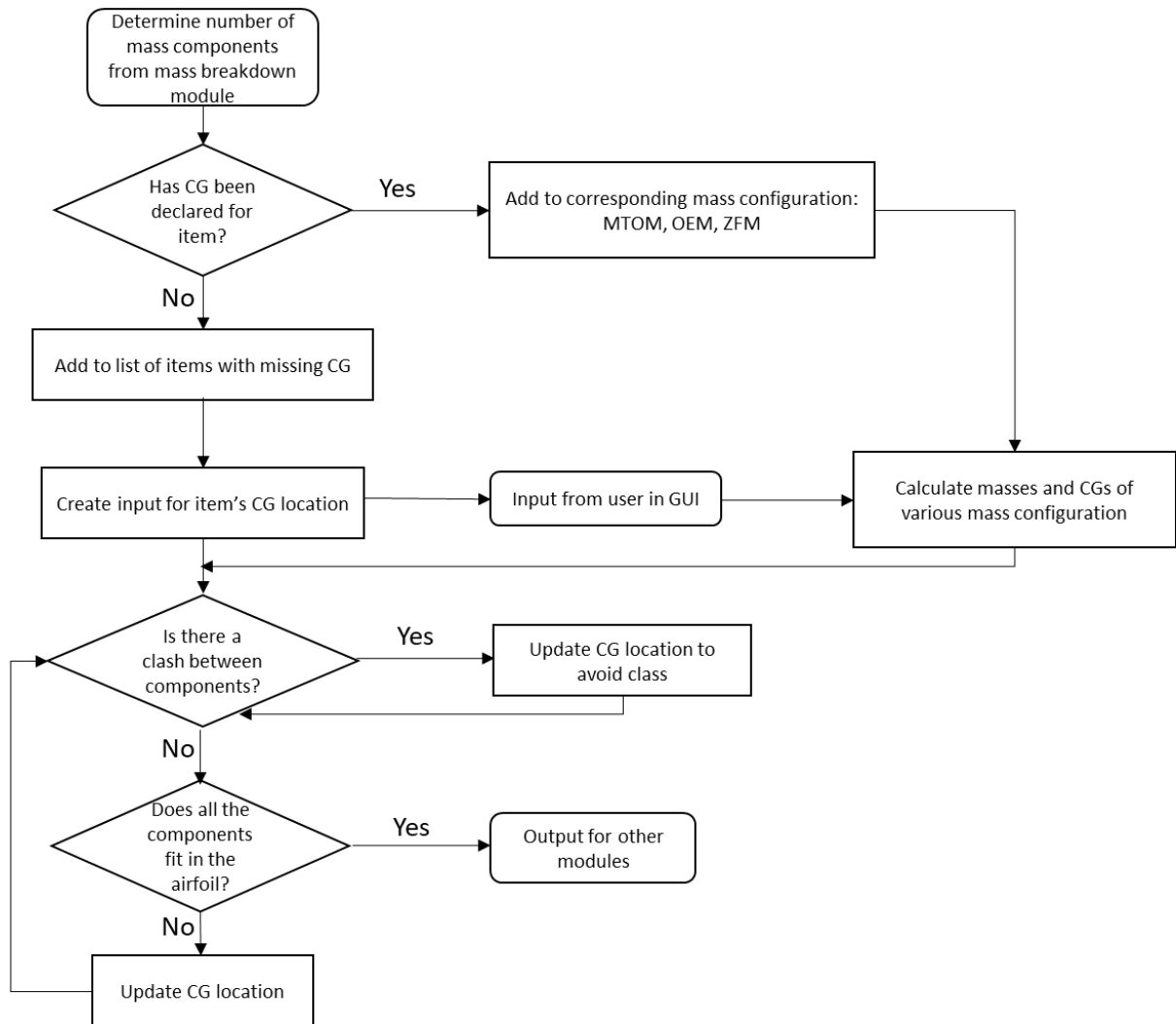


Figure 3-20 Routine for location items and resolving packaging issues

3.8.1 Cabin Sizing

The BWB passenger cabin is sized in accordance with the Bradley approach [116]. Two approaches can be used to determine the total pressurised cabin space. The first approach calculates the cabin area by adding the areas of all the required places in the cabin. The second technique calculates the area based on the cabin dimensions specified by users. Eq. (3-31) and (3-32) define these two methods.

$$A_{cabin} = A_{seats} + A_{servs} + A_{waste} + A_{aisle} \quad (3-31)$$

$$A_{cabin} = (X_{LP} + X_{LW}) * \frac{WF}{2} \quad (3-32)$$

The total area required for the passenger seating A_{seats} is calculated as the sum of the areas required for each class of seats multiplied by the number of passengers using Eq. (3-33). Eq. (3-33) is used in conjunction with user-supplied variables from Table 3-5 to calculate the area per seat for each class.

$$A_{seat} = \frac{BAYW * PITCH}{NABR} \quad (3-33)$$

where

A_{seat} = Area per seat, m^2

$BAYW$ = Passenger bay width, m

$PITCH$ = Seat pitch, m

$NABR$ = Number of seats abreast

Table 3-5 Default values for seat area

	Units	First Class	Business Class	Tourist Class
$NABR$	-	4	5	6
$PITCH$	m	1.55	1.00	0.81
$BAYW$	m	3.65	3.65	3.65

$$A_{seats} = NPT * A_{Tseat} + NPB * A_{Bseat} + NPF * A_{Fseats} \quad (3-34)$$

where

A_{seats} = Area for passengers, m^2

NPT = Number of tourist class passengers

A_{Tseat} = Seat area for tourist class passengers, m^2

NPB = Number of business class passengers

A_{Bseat} = Seat area for business class passengers, m^2

NPF = Number of first-class passengers

A_{Fseats} = Seat area for first class passengers, m^2

The utility area is made up of lavatories, galleys, and closets. The following equations can be used to estimate these areas.

Lavatory Area: The number of lavatories is estimated using the following equation.

$$N_{lava} = \left\lfloor 0.99 + \frac{NPF}{16} \right\rfloor + \left\lfloor 0.99 + \frac{NPB}{24} \right\rfloor + \left\lfloor 0.99 + \frac{NPT}{40} \right\rfloor \quad (3-35)$$

where

$\left\lfloor 0.99 + \frac{NPF}{16} \right\rfloor$ = Floor function of $0.99 + \frac{NPF}{16}$. Equal to the nearest integer rounded down.

The area for each lavatory including half of an aisle is estimated with the following equation.

$$A_{lava} = \frac{BAYW * 3.0}{2.0} \quad (3-36)$$

Galley Area: The number of galleys is estimated using the following equation.

$$N_{gally} = \lfloor 0.99 + 0.6 * N_{lava} \rfloor \quad (3-37)$$

The area for each galley including half of an aisle is estimated with the following equation.

$$A_{gally} = \frac{BAYW}{2.0} \quad (3-38)$$

Closet Area: The number of closets is estimated using the following equation.

$$N_{closet} = \lfloor 0.99 + 0.4 * N_{lava} \rfloor \quad (3-39)$$

The area for each closet including half of an aisle is estimated with the following equation.

$$A_{closet} = \frac{BAYW}{2.0} \quad (3-40)$$

Total Utility Area: The total area required for the utilities is estimated with the following equation.

$$A_{servs} = N_{lava} * A_{lava} + N_{gally} * A_{gally} + N_{closet} * A_{closet} \quad (3-41)$$

The area of the cabin that is wasted due to the slanted side wall is estimated with the following equation.

$$A_{waste} = \frac{NBAY * \tan LE * BAYW^2}{4.0} \quad (3-42)$$

where

$NBAY$ = Number of passenger bays. User input or estimated using Eq. (3-43).

$\tan LE$ = Tangent of the cabin leading edge sweep angle

The number of passenger bays for a BWB aircraft is defined as:

$$NBAY = \left\lceil 0.5 + \frac{A_{seats} + A_{servs}}{550} \right\rceil \quad (3-43)$$

The BWB fuselage width is estimated using the equation below

$$WF = NBAY * BAYW \quad (3-44)$$

The transverse aisles area from centre to centre of outboard bays is given below.

$$A_{AISLE} = 10.0 \times (NBAY - 1) \times BAYW \quad (3-45)$$

3.8.2 Wing Tank Volume

Wing tank volume calculations are performed with differential volumes, as shown in Figure 3-21, and Eq. (3-46) and (3-47). The forward and aft wing tank limits are imposed by user as the assumed wing spar positions.

$$\Delta v = \frac{\Delta y}{6} [\Delta x_1 (2\Delta z_1 + \Delta z_2) + \Delta x_2 (\Delta z_1 + 2\Delta z_2)] \quad (3-46)$$

$$\Delta z_j = \frac{1}{2} (z'_j + z''_j) \quad (3-47)$$

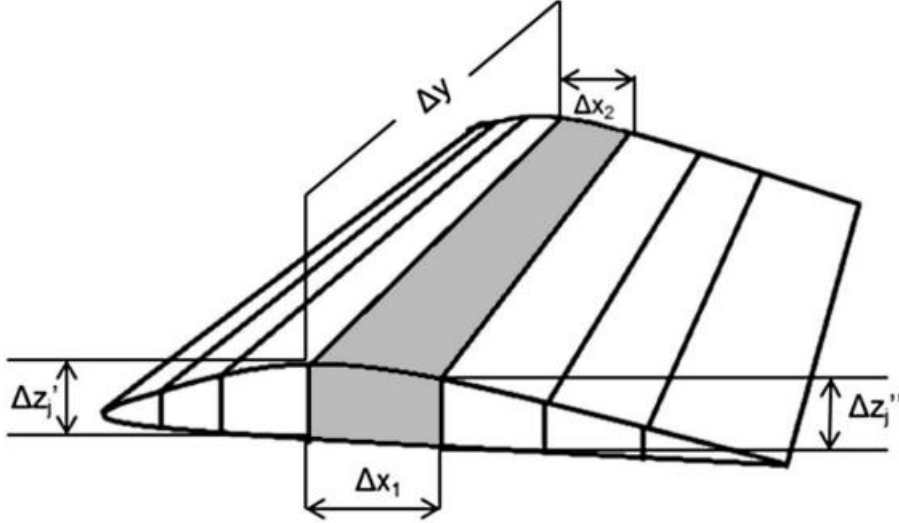


Figure 3-21 Differential volumes in a wing section

The total fuel volume is obtained as the sum of the differential volumes, multiplied by a correction factor less than one which accounts for the structural density, as shown in Eq. (3-48)

$$Vol_{fuel} = f_{structure} \sum_{i=1}^n \Delta v_i \quad (3-48)$$

This procedure can be applied to any area of a tailless configuration in order to represent complex, distributed fuel tank systems. The total fuel centre of gravity can then be obtained as:

$$X_{CG_{fuel}} = \sum_{i=1}^n \frac{\rho_{fuel} * \Delta v_i * x_{CG_{fuel_i}}}{\rho_{fuel} * Vol_{fuel}} \quad (3-49)$$

3.9 Stability and Control

The stability and control module aims to assess the stability characteristics and trim capability of the aircraft for all user-defined flight conditions. In this research,

the longitudinal static stability of a BWB is constrained through the static margins and control surface deflections required for achieving trim at all flight conditions.

The longitudinal static margin (K_n) is a function of the CG position which is from the packaging and CG module and the neutral point from the aerodynamic analysis module. The equation for static margin is given by Eq. (3-50).

$$K_n = \frac{X_{np} - X_{cg}}{\bar{c}} = -\frac{C_{m_\alpha}}{C_{L_\alpha}} \quad (3-50)$$

The neutral point can be estimated from the geometry and the aerodynamic data as:

$$X_{np} = X_{ref} - \bar{c} \frac{C_{m_\alpha}}{C_{L_\alpha}} \quad (3-51)$$

Trim deflection is obtained as the deflection angle of some control effector for which the sum of moments refer the aircraft's CG is zero, as given in Eq. (3-52). For the case of longitudinal trim, the moment axis is the lateral or Y axis.

$$\sum M_{y_{GG}} = \left(\frac{dC_{M_{ac}}}{d\delta_e} \right) \delta_e - \frac{x_w}{\bar{c}} \left[\left(\frac{dC_L}{d\alpha} \right) \alpha + \left(\frac{dC_L}{d\delta_e} \right) \delta_e \right] = 0 \quad (3-52)$$

In the GENUS, Digital DATCOM [48] has been implemented as the stability and control analysis tool, which is also suitable for BWB aircraft. DATCOM can calculate the effect of symmetrical control effectors on lift and moment coefficients automatically. The trim angle at each flight condition can be calculated based on the control effector deflections and changes in lift and moment coefficients. Using the angle of attack and the trim deflection, the trim drag is estimated. The trim drag is found by considering the derivative of $C_{D_{\delta_e}}$ at a given α . Multiplying this derivative by the trim deflection δ_e results in the increase in drag coefficient due to the elevator deflection. According to [128], the trim drag only accounts for a small part (<1%) of the aircraft drag, and has little correlation with the shape change of the BWB aircraft, so in the subsequent drag analysis of the BWB aircraft, the trim drag is not listed separately.

3.10 Boundary Layer Ingestion

3.10.1 Boundary Layer Modelling

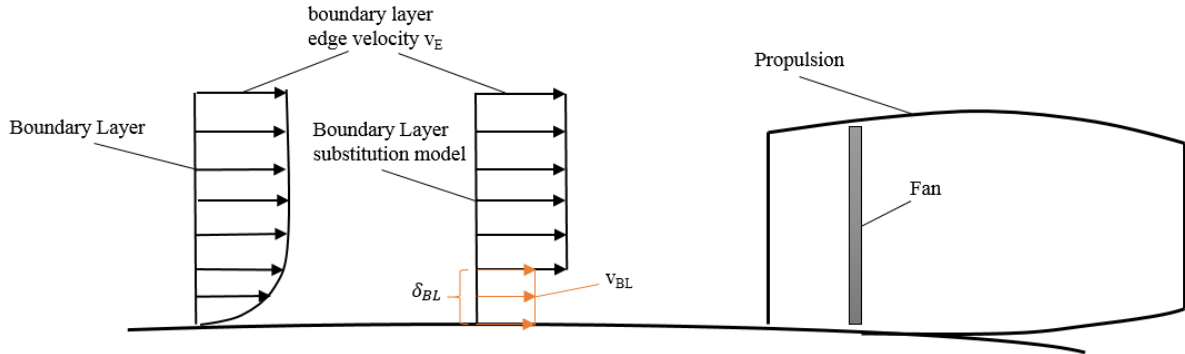


Figure 3-22 A diagram of the boundary layer

A diagram of the boundary layer is shown in Figure 3-22. According to the literature review, the CFD method with the Navier-Stokes equations for viscous flows can be used to accurately describe the flow characteristics in the boundary layer. However, CFD solvers are difficult to implement and require high computational cost when used in a multi-variable design and optimisation tool. At the conceptual design phase, the speed is the essential requirement rather than accuracy. Therefore, the Navier-Stokes equations can be reduced to a more tractable form by making some appropriate assumptions and applying the velocity profile approximation[46]. In this study, the $1/n^{th}$ power-law velocity profile approximation is applied to represent the velocity profile in a fully developed boundary layer [129]. The thickness δ and features of the boundary layer can also be predicted by a variety of methods. A modified boundary layer approximation for turbulent flat plate [130] is selected to estimate boundary layer thickness, as demonstrated in Eq. (3-53).

$$\delta = \frac{0.37x}{Re_x^{1/6}} \quad (3-53)$$

The displacement thickness (δ^*), momentum thickness (θ), and energy thickness (θ^*) are three additional dimensions that constitute the boundary layer [131], which can be defined:

$$\frac{\delta^*}{\delta} = \int_0^\infty \left[1 - \frac{\rho_y u_y}{\rho_0 u_0} \right] dy \quad (3-54)$$

$$\frac{\theta}{\delta} = \int_0^\infty \frac{\rho_y u_y}{\rho_0 u_0} \left[1 - \frac{u_y}{u_0} \right] dy \quad (3-55)$$

$$\frac{\theta^*}{\delta} = \int_0^\infty \frac{\rho_y u_y}{\rho_0 u_0} \left[1 - \left(\frac{u_y}{u_0} \right)^2 \right] dy \quad (3-56)$$

Each boundary layer thickness term represents the normal distance to a reference plane representing the lower edge of a hypothetical inviscid fluid of uniform velocity that has the same flow rate flow, momentum flow rate or kinetic energy as occurs in the real fluid with the boundary layer. The boundary layer flow characteristics can be represented as non-dimensional parameters using these definitions [132]:

$$MAG = \frac{\dot{m}_{BL}}{\rho_0 u_0 \delta w} = \int_0^1 \frac{\rho_y u_y}{\rho_0 u_0} d\left(\frac{y}{\delta}\right) = 1 - \frac{\delta^*}{\delta} \quad (3-57)$$

$$MOG = \frac{mom_{BL}}{\rho_0 u_0^2 \delta w} = \int_0^1 \frac{\rho_y}{\rho_0} \left(\frac{u_y}{u_0} \right)^2 d\left(\frac{y}{\delta}\right) = 1 - \frac{\delta^*}{\delta} - \frac{\theta}{\delta} \quad (3-58)$$

$$KEG = \frac{KE_{BL}}{0.5 \rho_0 u_0^3 \delta w} = \int_0^1 \frac{\rho_y}{\rho_0} \left(\frac{u_y}{u_0} \right)^3 d\left(\frac{y}{\delta}\right) = 1 - \frac{\delta^*}{\delta} - \frac{\theta^*}{\delta} \quad (3-59)$$

The mass flow group (MAG), the momentum group (MOG), and the kinetic energy group (KEG) are the non-dimensional parameters represented in Eq. (3-57) to (3-59). These groups describe boundary layer flow characteristics in a unit area relative to the free flow characteristics. Each of the boundary layer characteristics, $\dot{m}_{BL}$, mom_{BL} and KE_{BL} has a deficit compared to the free flow characteristics. Similarly, boundary layer also has a certain amount of total pressure deficit with respect to the free flow, which can be defined as a mass-averaged value:

$$\frac{\bar{P}_{BL}}{P_0} = \frac{1}{\dot{m}_{BL}} \int_{BL} \frac{P_y}{P_0} d\dot{m} = \frac{1}{MAG} \int_0^1 \frac{P_y}{P_0} \frac{u_y}{u_0} \frac{\rho_y}{\rho_0} d\left(\frac{y}{\delta}\right) \quad (3-60)$$

In a similar integral way, the average velocity of the entire boundary layer flow can be given:

$$\frac{\bar{u}_{BL}}{u_0} = \frac{1}{\dot{m}_{BL}} \int_{BL} \frac{u_y}{u_0} d\dot{m} = \frac{1}{MAG} \int_0^1 \frac{u_y^2 \rho_y}{u_0^2 \rho_0} d\left(\frac{y}{\delta}\right) \quad (3-61)$$

In addition, some general definitions can be used to determine the boundary layer properties. When assuming a power-law velocity profile is applied:

$$\frac{u_y}{u_0} = \left(\frac{y}{\delta}\right)^{\frac{1}{n}} \quad (3-62)$$

$$\frac{\rho_y}{\rho_0} = \left[1 + \frac{\gamma-1}{2} Ma^2 \left(1 - \left(\frac{y}{\delta}\right)^{\frac{2}{n}}\right)\right]^{-1} \quad (3-63)$$

$$\frac{P_y}{P_0} = \left[1 + \frac{\gamma-1}{2} Ma^2 \left(1 - \left(\frac{y}{\delta}\right)^{\frac{2}{n}}\right)\right]^{\frac{\gamma}{\gamma-1}} \quad (3-64)$$

This major objective of this step is to estimate the boundary layer characteristics: mass flow, average total pressure, and average velocity. Any approaches for obtaining the boundary layer characteristics can be used, as long as the resulting boundary layer characteristics meet the requirements.

3.10.2 Inlet Model

An ingested turbulent boundary layer can result in non-uniform total pressure and swirl distributions, which can have a detrimental effect on an engine's performance. The distortion in the intake can result in a greater total pressure loss in the inlet than with a laminar free-stream flow engine. In rare instances, the unfavourable pressure gradient created by an intake may result in the separation of the boundary layer. Boundary layer ingesting systems are frequently used in conjunction with S-duct intakes to facilitate integration into the airframe.

Previous study has demonstrated that the effectiveness of BLI propulsors is highly dependent on the total pressure loss at the inlet [94]. Total pressure loss results in a decrease in the effective pressure ratio. As a result, the propulsion system must consume additional mass flow to provide the required thrust.

Consuming more free-stream mass flow results in a shift in the inlet flow characteristics toward free-stream, hence diminishing the BLI benefits. Thus, the inlet model is a critical component of the BLI analysis approach since it allows for the prediction of total inlet recovery under distorted flow conditions.

In this study, the overall pressure loss in the inlet duct that ingests the boundary layer flow is predicted with Seddon's method [133]. The method calculates the total pressure loss as a function of the area of the intake and inlet streams and the capture ratio of the boundary layer. While boundary layer control techniques may be able to lower total pressure loss through the inlet, the method only provides an estimate of total pressure loss. This can be approximated as follows for subsonic flow:

$$\frac{\Delta P_1}{P_0} = QK\Phi \quad (3-65)$$

$$Q = 1 - \left(\frac{A_i}{A_{ff}} \right)^3 \quad (3-66)$$

$$K = \frac{\theta}{h} \frac{A_1}{A_{ff}} \left[0.6 + 2.4 \left(\frac{A_m}{A_1} - 1 \right) \right] \quad (3-67)$$

Where:

Φ is an empirical the boundary layer function shown in Figure 3-23.

A_m is the area of the midpoint between the highlight and the fan face.

A_{ff} is the fan face area.

A_1 is the highlight area.

For subsonic Mach numbers, Φ can be approximated by the following polynomial:

$$\Phi = 5.045Ma_i^3 - 4.753Ma_i^2 + 1.878Ma_i - 0.276 \quad 0.3 < Ma_i < 1.0 \quad (3-68)$$

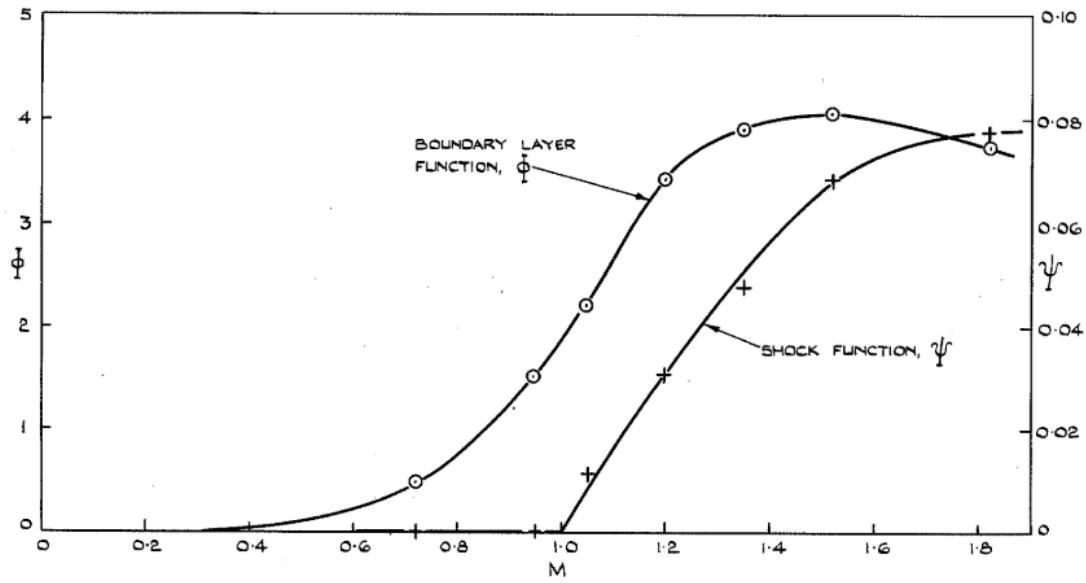


Figure 3-23 Boundary layer function vs Mach number [133]

Assuming the duct area is a straight taper from the highlight to the fan face, the total pressure recovery via the duct can be calculated by:

$$\eta_1 = 1 - \Phi \frac{\theta}{h} \left(1.2 - 0.6 \frac{A_1}{A_{ff}} \right) \left[1 - \left(\frac{A_i}{A_{ff}} \right)^3 \right] \quad (3-69)$$

Where:

A_i is the cross-sectional area of the streamtube at station i .

Due to the complex aerodynamics of serpentine inlets, the detailed design is out of the scope of this research. The inlet size is a function of freestream Mach number, the assumed Mach number at the throat, and the maximum allowable Mach number at the engine front face. Throughout this research, the contraction ratio between the inlet duct's highlight and throat of 0.9, a maximum throat Mach number of 0.75 and a maximum engine front face Mach number of 0.67 have been assumed, as shown in. Using isentropic flow relation, Eq.(3-70), the inlet capture area is given by Eq.(3-71).

$$\frac{A}{A^*} = \frac{1}{Ma} \left[\frac{1 + 0.2Ma^2}{1.2} \right]^3 \quad (3-70)$$

$$\frac{A_i}{A_{ff}} = \left(\frac{A}{A^*} \right)_i / \left(\frac{A}{A^*} \right)_{th} * \left(\frac{A}{A^*} \right)_{th} / \left(\frac{A}{A^*} \right)_{ff} \quad (3-71)$$

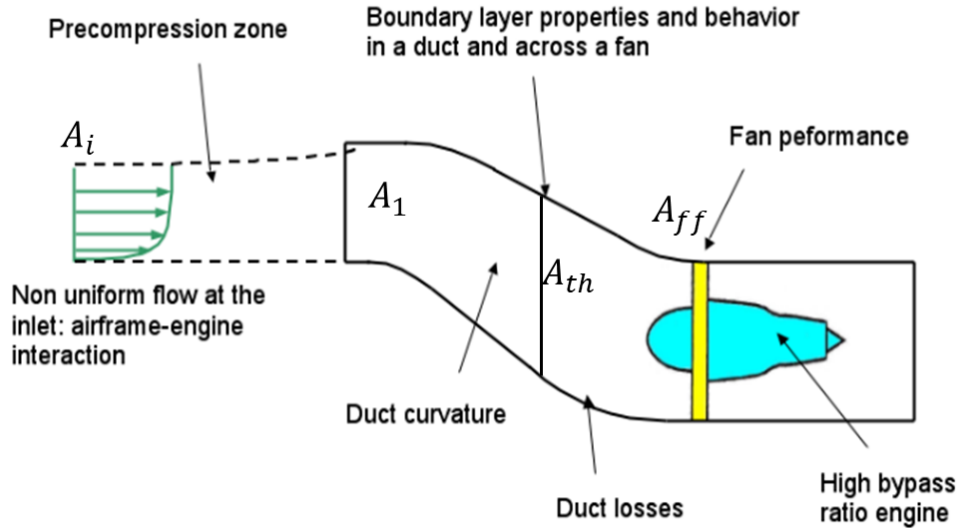


Figure 3-24 Inlet duct schematic diagram

To ensure that realistic estimates for η_1 are provided, it is appropriate to assume that $\Phi = 0$ when $Mach < 0.30$ and that η_1 will never exceed one.

3.10.3 Inlet Characteristics for BLI System

The intake may be more or less than the full boundary layer flow due to the different flow rates required by the engine. Therefore, a step in the process is required to compare the height of the inlet stream with the thickness of the boundary layer to determine the inlet flow characteristics of the engine. Assuming a rectangular streamtube with a width w and a height h , the following three conditions are possible:

1. Ingest a fraction of the boundary layer ($h/\delta < 1$).
2. Ingest entire boundary layer and no free-stream ($h/\delta = 1$).
3. Ingest entire boundary layer and free-stream ($h/\delta > 1$).

For the first condition, where the inlet ingests only a fraction of the boundary layer, the upper limit of Eq. (3-57) to (3-59) should become h/δ , the ratio of streamtube height to boundary layer thickness. In the second condition, the characteristics of the inlet flow can be estimated with the equations presented in the preceding section. In the third condition, ingesting flow characteristics must incorporate boundary layer flow and free-stream. The features and profile of the boundary layer can be determined in the conventional manner. The total mass flow rate of the input stream becomes the sum of boundary layer mass flow rate and the ingested free-stream mass flow rate, which can be defined as:

$$\frac{\dot{m}_{total}}{\rho_0 u_0 \delta w} = \frac{\dot{m}_{BL} + \dot{m}_{FS}}{\rho_0 u_0 \delta w} = \frac{h}{\delta} + MAG - 1 \quad (3-72)$$

The total pressure deficit of both free-stream flow and the boundary layer, $\bar{P}_i$, is the mass-averaged value obtained by integrating over the full inlet flow, which can be expressed in the non-dimensional form of Eq. (3-73).

$$\frac{\bar{P}_i}{P_0} = \frac{1}{\dot{m}_{total}} \left[\int_0^\delta \frac{P_y}{P_0} d\dot{m} + \int_\delta^h d\dot{m} \right] = \frac{\frac{h}{\delta} - 1 + \frac{\bar{P}_{BL}}{P_0} \cdot MAG}{\frac{h}{\delta} + MAG - 1} \quad (3-73)$$

The average velocity $\bar{u}_i$ of the inlet flow can be computed similarly by integrating over the full inlet flow (Eq. (3-74)).

$$\frac{\bar{u}_i}{u_0} = \frac{\frac{h}{\delta} - 1 + \frac{\bar{u}_{BL}}{u_0} \cdot MAG}{\frac{h}{\delta} + MAG - 1} \quad (3-74)$$

3.10.4 BLI Benefits Quantification

The benefits of BLI to the airframe can be quantified in terms of the ram drag recovered from the BLI propulsion system, as demonstrated in References [134][104][135]. Drela [136] proposed a power balance method to present the BLI benefit from an aerodynamic perspective based on a particular control volume analysis. According to power balance method, the estimate of the power balance reduction brought by BLI system is expressed by:

$$\beta \phi_p^* = (1 - \nu) K_\infty b \quad (3-75)$$

Where:

β is the percentage of the wake which is recovered by BLI.

ϕ_p^* is wake dissipation factor.

ν is the fraction of that defect which remains after ingestion.

K_∞ is the total wake defect at the trefftz plane.

b is the width of the stream-tube prior to the pre-entry zone, which is assumed to be equal to the width of the fan.

The turbulent closure relations for a free-wake region are used to convert the edge velocity and BL properties (δ^* , θ) into the kinetic energy defect θ^* , and the value of K_∞ is computed from its definition (Eq. (3-76)).

$$K = \frac{1}{2} \rho_e u_e^3 \theta^* \quad (3-76)$$

The parameter ν is computed by estimating the ratio of the upper surface wake to the lower wake at the trailing edge:

$$\nu = \left(\frac{K_L}{K_L + K_U} \right)_{TE} = \frac{\theta_L^*}{\theta_L^* + \theta_U^*} \quad (3-77)$$

Subscript U , L and ∞ represent the upper surface, lower surface and the free stream respectively.

Therefore, ν effectively represents the proportion of the profile drag which comes from the lower surface and $(1 - \nu)$ is the upper surface proportion. Then, the recovered drag D_{BLI} per unit length can be obtained by Eq. (3-78).

$$\frac{D_{BLI}}{b} = \frac{\beta \phi_p^*}{u_\infty} = (1 - \nu) \frac{K_\infty}{u_\infty} = \frac{1}{2} (1 - \nu) \rho \theta_\infty^* u_\infty^2 \quad (3-78)$$

All required parameters can be obtained in the previous section on boundary layer modelling.

3.10.5 Fan Model

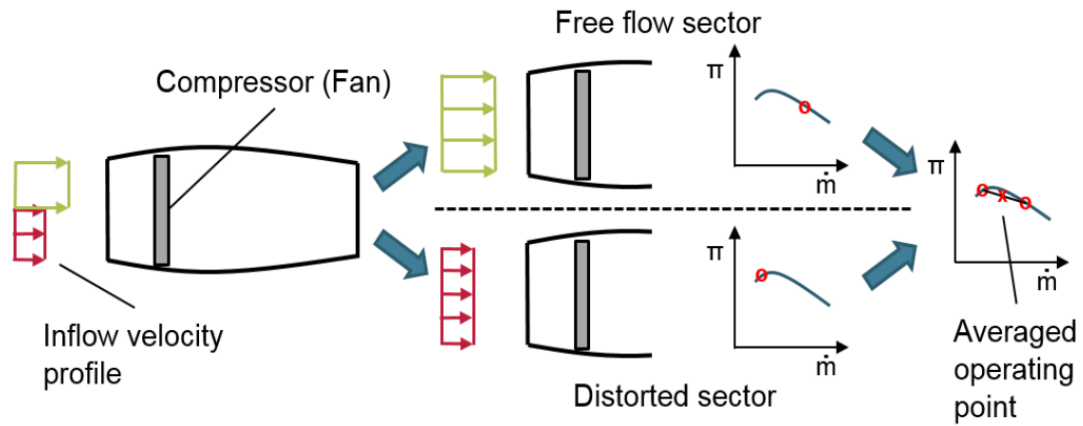


Figure 3-25 Schematic diagram of parallel compressor model

The effect of intake of low-speed fluids on fan performance is analysed using a parallel compressor model. The basic principle of the parallel compressor model is to replace the compressor that exhibits inflow distortion with two identical hypothetical compressors operating at different operating points. The two sub-compressors' operational characteristics are identical to the undistorted original characteristic. Once the two operating points for the undistorted and distorted states have been obtained, an area-weighted average value must be determined in order to determine the individual averaged operating point, which shown in Figure 3-25. The detailed calculation steps are demonstrated in Chapter 2.3.4.3.2.

A mass flow matching process links the BLI model to the engine analysis tool EngineSim. The procedure of this BLI analysis method is as follows (Figure 3-26):

1. Establish the boundary layer properties:
 - Reynolds number Re
 - Boundary layer thickness δ
 - Local free-stream velocity u_0
2. Determine the boundary layer flow characteristics

- Boundary layer mass flow $\dot{m}_{BL}$
 - Average total pressure $\bar{P}_{BL}$
 - Average velocity $\bar{u}_{BL}$
3. Guess streamtube height h and hence obtain h/δ
 4. Determine inlet flow characteristics
 - Streamtube mass flow $\dot{m}_{total}$
 - Average total pressure $\bar{P}_i$
 - Average velocity $\bar{u}_i$
 5. Determine the fan efficiency η_{fan} using parallel compressor model
 6. Simulate the propulsion performance by the EngineSim
 - Engine mass flow $\dot{m}_{engine}$
 - Thrust and SFC
 7. Return to Step 3 until the streamtube mass flow $\dot{m}_{total}$ in Step 4 matches the engine mass flow demand $\dot{m}_{engine}$ in Step 6
 8. Transfer the propulsion system simulation results to the performance module

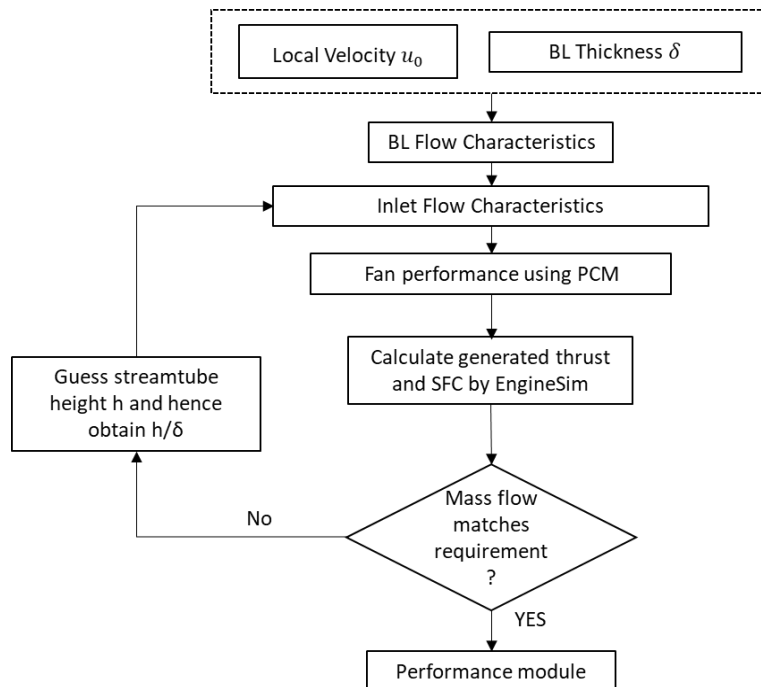


Figure 3-26 Process structure of the BLI analysis module

3.10.6 BLI model Validation

A validation procedure was used to determine the model's applicability. Gladin [104] previously simulated a high-bypass turbofan with BLI and utilised the results to conduct a preliminary validation.

The baseline vehicle used here is very similar to the Boeing N2A-EXTE design. The vehicle is intended to carry 300 passengers and would therefore be a future potential replacement for a Boeing 777 (double aisle) type airplane. Some overall assumed parameters for the vehicle which are relevant to the BLI problem are listed in Table 3-6 and the vehicle planform geometry is shown in Figure 3-27.

Table 3-6 Key parameters for the baseline BWB vehicle [104]

Parameter	Value
Wing Span (m)	65
Total Length (m)	45
Cruise Mach	0.8
Cruise Altitude (m)	10668

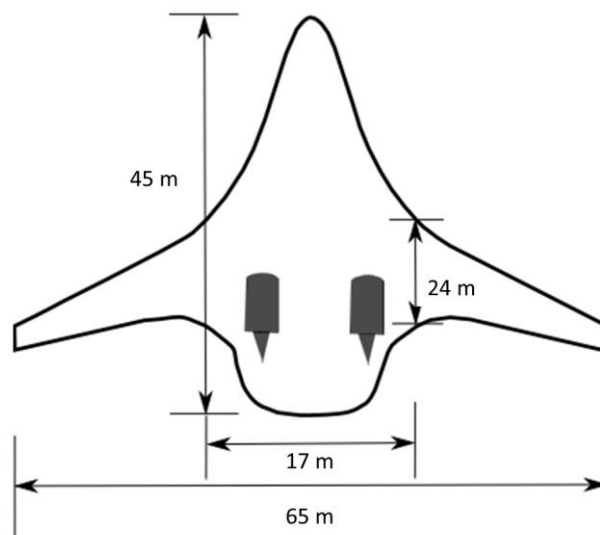


Figure 3-27 BWB baseline design dimensions

Boundary Layer Modelling

The flow profile on this baseline BWB design is extracted from previous research [137]. This research presents CFD results for boundary layer profiles around the

rear of the airframe at different locations on the centreline of the N2A fuselage, ranging from 60% chord to 100% chord. The boundary layer properties(δ , δ^* and θ) and edge velocity U_e have been computed from these CFD results by Gladin [104] and compared to an equivalent analysis by the boundary layer modelling method developed in this research. The comparison results are shown in Figure 3-28 and it is clear that the method developed in this research provides reasonable estimates for the vehicle boundary layer trend.

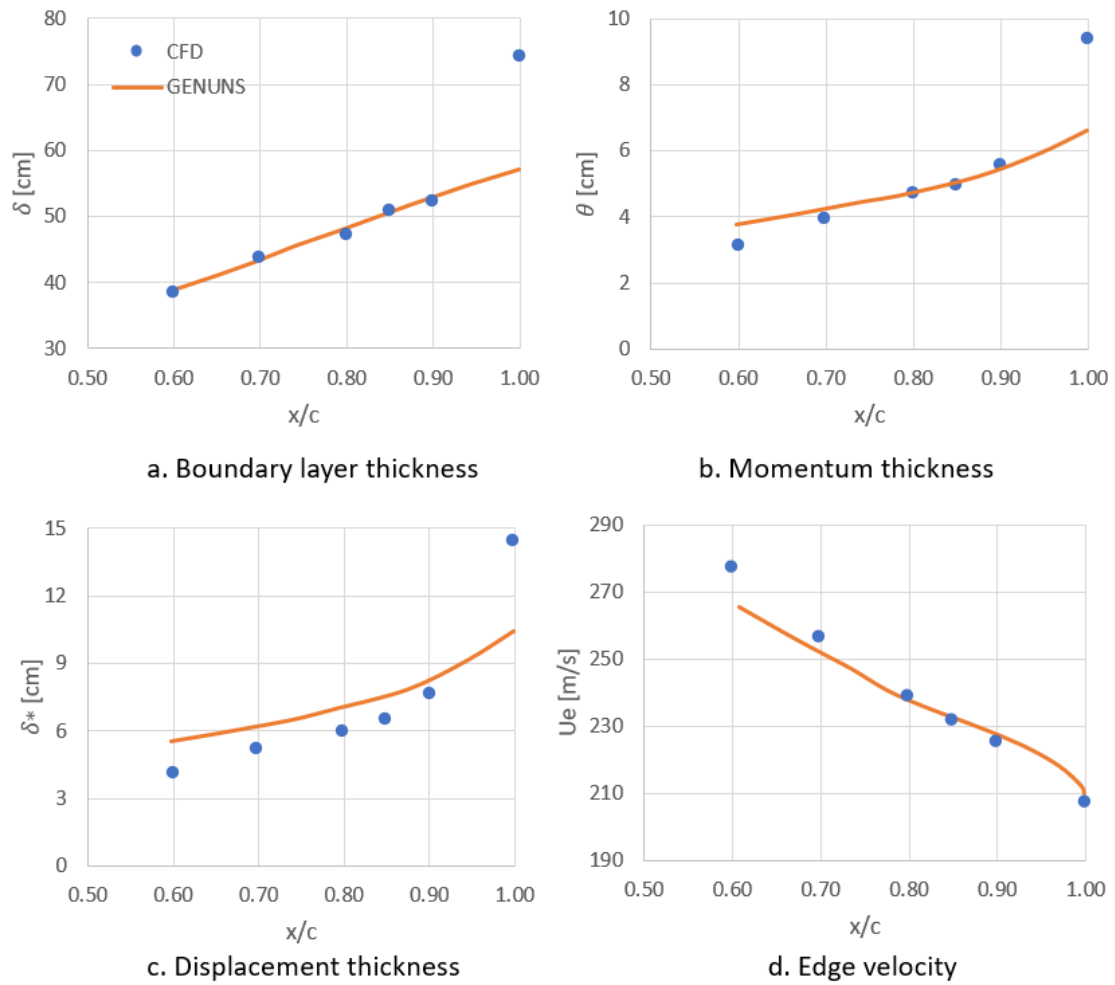


Figure 3-28 Comparison of boundary layer parameters

Inlet Characteristics

The reference [137] provided the mean total pressure and Mach number difference relative to the free-stream for input streams of various heights derived

from CFD results. The inlet flow characteristics method presented in this study was used to simulate the flow characteristics over the reference's range of streamtube heights. The validation findings are depicted in Figure 3-29. When the intake stream parameters are compared to the CFD findings, an average difference in boundary layer total pressure loss of 5.4% and that in Mach number of 0.13 are observed. This illustrates that the analytically anticipated input stream parameters closely match the CFD-derived results.

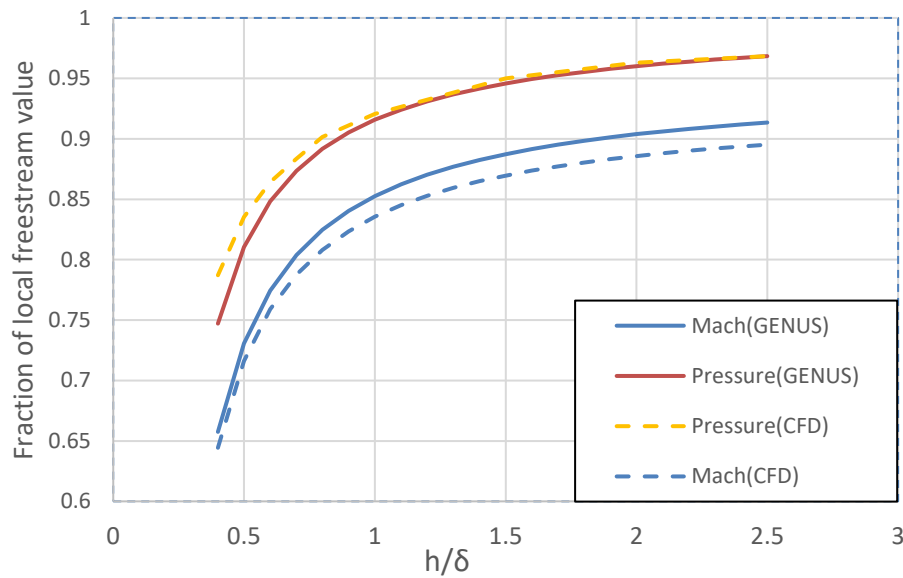


Figure 3-29 Comparison of the inlet flow characteristics against CFD data

Propulsion simulation model

The high-bypass ratio turbofan engine on this BWB baseline design has been simulated by the modified EngineSim with BLI model, which has integrated in the GENUS. The results are compared to the results from Gladin [104].

The baseline propulsion system architecture is a typical high-bypass ratio turbofan. The baseline technology parameters and design variables are listed in table. The primary cycle design variables for the system are the overall pressure ratio and fan pressure ratio. The compressor pressure ratio is varied to match the overall pressure ratio of 42.8, which is selected to maintain reasonable T3 values.

Table 3-7 Propulsion system technology parameters and design variables

Parameter	Value/Range
Fan Pressure Ratio	1.25-1.65
Overall Pressure Ratio	42.8
Maximum Turbine Inlet Temperature T4	3450R
Maximum Compressor Exit Temperature T3	1800R
Fan Efficiency	0.9345
Compressor Efficiency	0.88
Burner Efficiency	0.997
High Pressure Turbine Efficiency	0.9251
Low Pressure Turbine Efficiency	0.9376
Cruise T4	2963R
Total Cruise Thrust Requirement	108.9kN

Typical power hook characteristics for the engine generated by the GENUS are shown in Figure 3-30, which is compared to the result from reference [104] . The results indicate the similar trend.

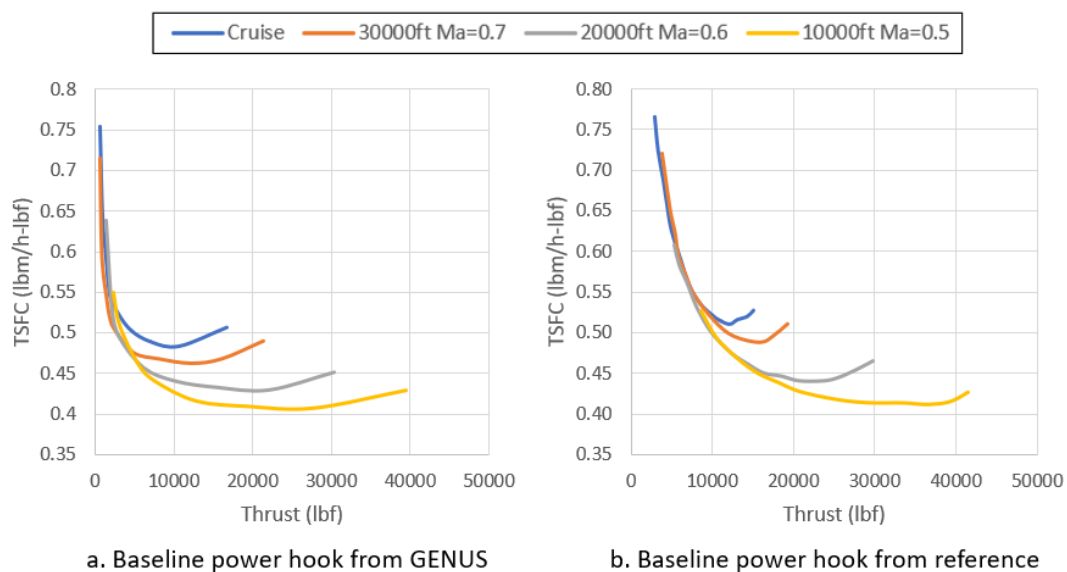


Figure 3-30 Comparison of baseline power hook

In order to evaluate the benefit of the BLI, the baseline engine are simulated in three conditions:

Condition 1: the baseline model with no BLI effects.

Condition 2: the baseline model with BLI, but no loss due to the BLI.

Condition 3: the baseline model with BLI, considering inlet recovery loss and fan loss.

These simulation for three conditions are all conducted at cruise condition. Figure 3-31 shows the TSFC for a range of fan pressure ratio at three conditions. The interesting result is that the curve with the full parallel compressor model looks very similar to that of the original baseline model, which, to some degree is dominated by the balance of BLI driving to lower fan pressure ratios and inlet recovery loss driving it upward. The results are compared to the reference data [104] shown in fig. The comparison indicates that result from GENUS performs a similar trend with reference data and could be obtain the consistent conclusion, which helps to validate the model.

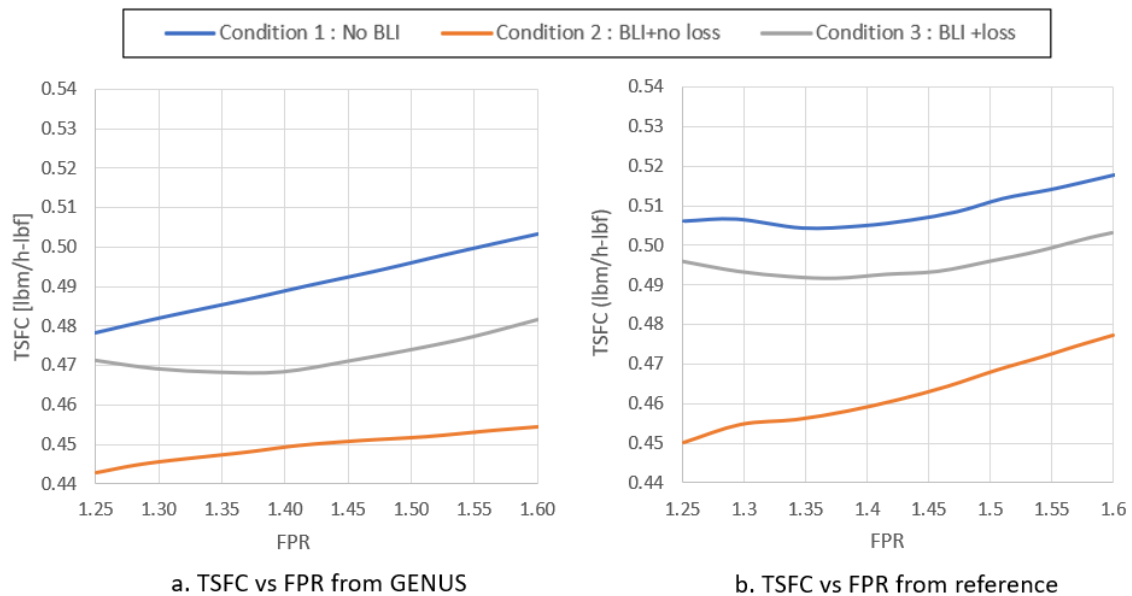


Figure 3-31 Comparison of TSFC for various fan pressure ratio

3.11 Framework Validation

The concept design of BWB is challenging since the correlations and methodologies used for tube-and-wing aircraft is not suitable for it. To accurately represent BWB aircraft, physics-based relationships are required. A robust and effective conceptual design methodology for BWB has been developed and implemented into the GENUS environment to evaluate the performance of BWB.

The concept BWB aircraft named SUGAR Ray, developed as part of NASA's Subsonic Ultra Green Aircraft Research (SUGAR) project [7], was generated in GENUS using publicly available data for the test case below. Its flight performance is simulated in GENUS for a 2,100-nautical-mile mission a cruise Mach number of 0.74. and a cruise altitude of 35,000 feet.

The design consists of a BWB geometry, which is shown in Figure 3-32. NACA4 aerofoils varying thicknesses are applied throughout design process, and the 18% and 10% thicknesses have been assumed for aerofoils at the centre and tip respectively, with a linear interpolation in between centre and the tip. The comparison of GENUS geometry and mission results and published data on the SUGAR Ray is shown in Table 3-8.

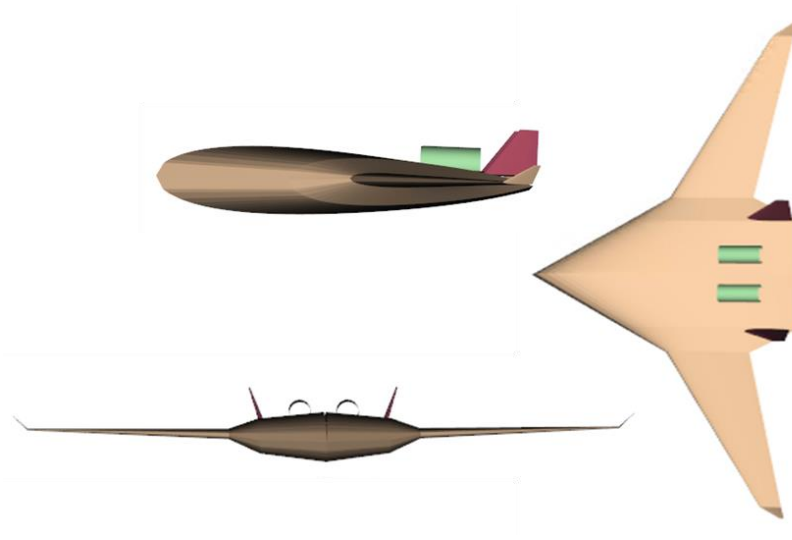


Figure 3-32 NASA SUGAR Ray 3-view drawing from GENUS

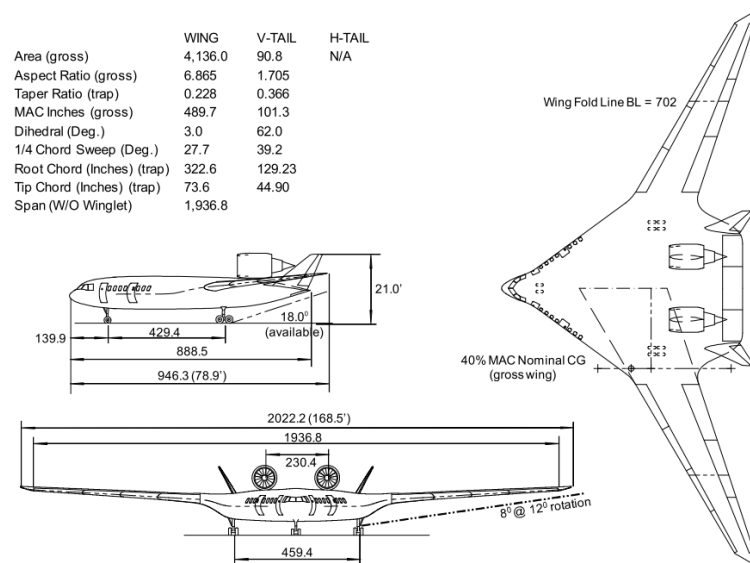


Figure 3-33 NASA SUGAR Ray configuration layout from Ref [7]

Table 3-8 SUGAR Ray configuration and mission variables from NASA and GENUS

Parameter	NASA	GENUS
Gross Wing Area (m ²)	384.25	398.54
Wingspan (m)	49.19	50.51
Gross Wing AR	6.87	6.50
Sweep Angle (degree)	27.7	27.7
Take-off Gross Weight (kg)	82766	83921
Passengers	155	155
Cruise Altitude (m)	10668	10668
Cruise Mach	0.74	0.74

The aerodynamic performance are evaluated by the panel method PANAIR in the GENUS. The representative aerodynamic results in the cruise condition are shown in Table 3-9 below.

Table 3-9 Comparison of GENUS aerodynamics results and published data on the SUGAR Ray

Result in Cruise	NASA	GENUS
L/D	26.60	26.01
CL	0.30	0.28
CD - Total Drag	0.01127	0.01074
CDf - Friction Drag	0.00596	0.00621
CDc - Wave Drag	0.00063	0.00060
CDi - Induced Drag	0.00468	0.00393

GENUS produced results that were substantially consistent with available data on the SUGAR Ray BWB. The obtained drag values were consistent with expectations, albeit with a lower compressibility drag and a larger friction drag in comparison to the overall cruise drag. In general, GENUS accurately simulates the SUGAR Ray's lift-to-drag ratio (L/D), returning a value of 26.01, compared to 26.6 in the NASA report.

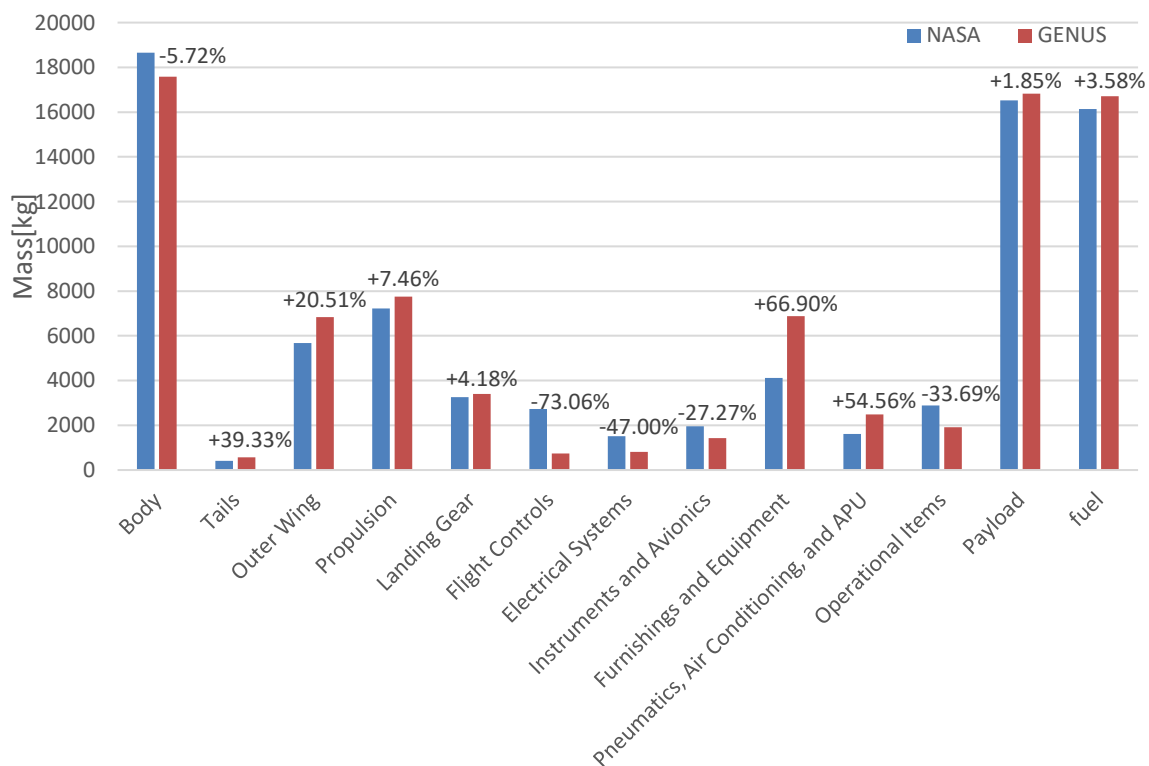


Figure 3-34 GENUS and NASA estimates of the SUGAR Ray mass breakdown

The SUGAR Ray aircraft's weight was estimated using the BWB weight estimation method described in Chapter 3.4, which is shown in Figure 3-34. The weight estimation methodologies integrated in the GENUS were effective in determining the structural weights of the SUGAR Ray's centerbody, outer wing, and landing gear to within 10% of the NASA report's results. The stated weights for flight controls, electrical systems, and avionics, on the other hand, were much more than those determined using GENUS. GENUS discovered that furnishings and equipment, as well as pneumatics, air conditioning, and the auxiliary power unit (APU), all weighed more. Despite these differences, there is general uniformity in the system weight.

Due to the lack of data in the NASA N+3 report regarding the process used to determine system weights, the exact source of the disparity is unknown. The following are some of the anticipated reasons:

1. GENUS and NASA may have different methods for estimating the weight of the flight controls. By default, the hydraulic and pneumatics in the main wing are calculated separately in the GENUS approach and the flight controls only include the tail actuation system. NASA's flight control results may include control actuators in the main wing, resulting in greater flight control weight and lower pneumatics weight.
2. The SUGAR Ray is built around a large number of electromechanical actuators, which result in the substantial weight of the electrical system.
3. The SUGAR Ray is an aircraft that can be put into service approximately 20 years in the future. Therefore, the weights of air conditioning, equipment, furnishings and the APU should be less than those calculated by GENUS using data from recent aircraft.

Overall, GENUS was able to accurately estimate the structural weights of the SUGAR Ray BWB configuration and GENUS was able to determine the operating empty weight to within 5% of published data.

In terms of propulsion, a conceptual engine named “gFan+” is installed on the SUGAR Ray. Its architectural concept is a 2-spool separate flow turbofan with a

59 OPR and 13 BPR (at top of climb), which is shown in Figure 3-35. The same engine has been modelled by the EngineSim in GENUS. The comparison of GENUS engine model and published data for the “gFan+” is listed in Table 3-10, which shows the similar results.

The payload-range diagram obtained from GENUS is also compared to the published data from NASA, which is shown in Figure 3-36.

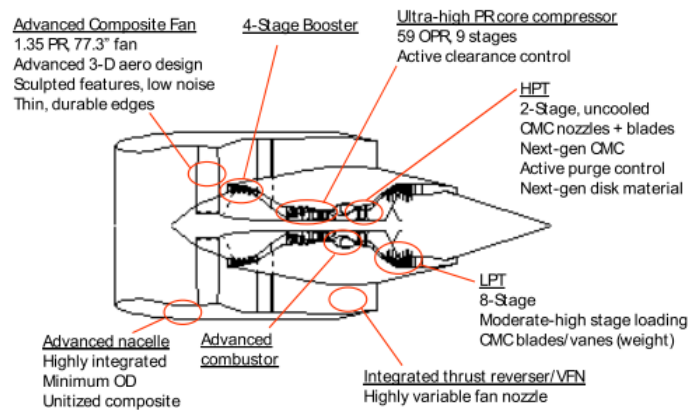


Figure 3-35 "gFAN+" Engine [7]

Table 3-10 Comparison of GENUS results and published data for "gFan+" engine

Parameter	gFan+	GENUS
Overall pressure ratio	59.0	59.0
Bypass ratio	13.0	13.0
Fan pressure ratio	1.35	1.35
Fan diameter (m)	1.96	1.96
Thrust at SLS (kN)	83.6	82.8
SFC at SLS (g/kN/s)	5.96	6.11
Thrust at cruise (kN)	13.5	12.9
SFC at cruise (g/kN/s)	13.27	13.53

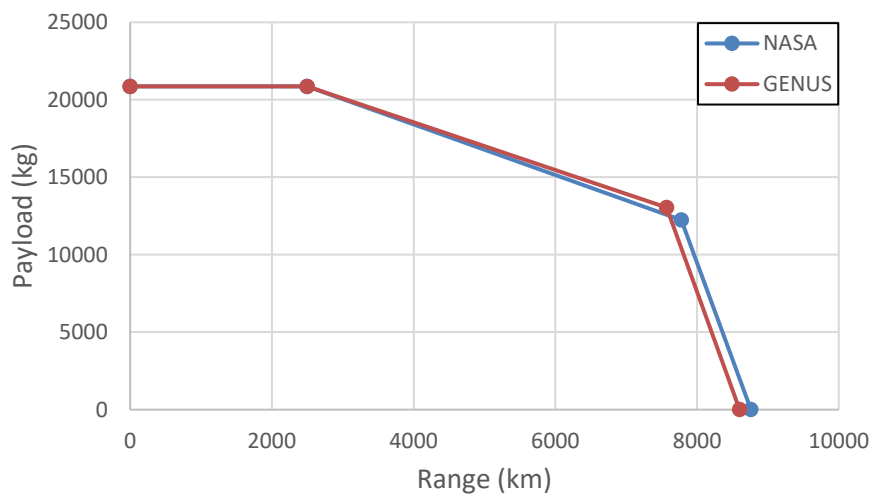
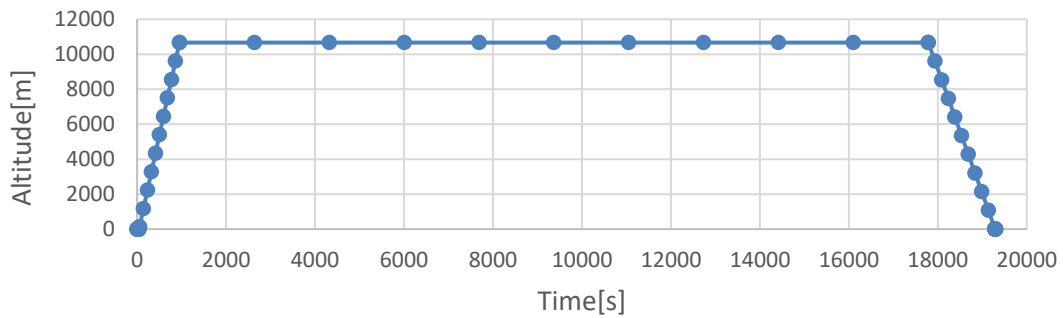
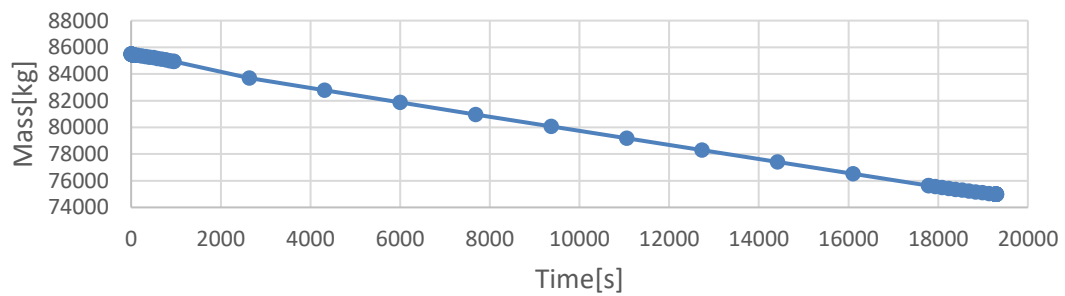


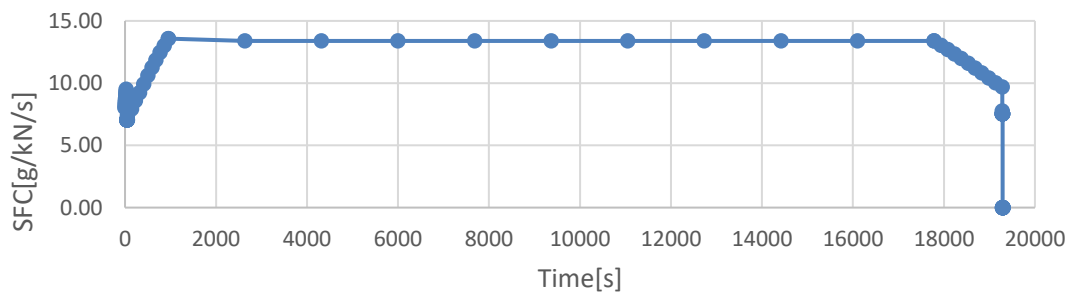
Figure 3-36 Comparison of the payload-range for SUGAR Ray



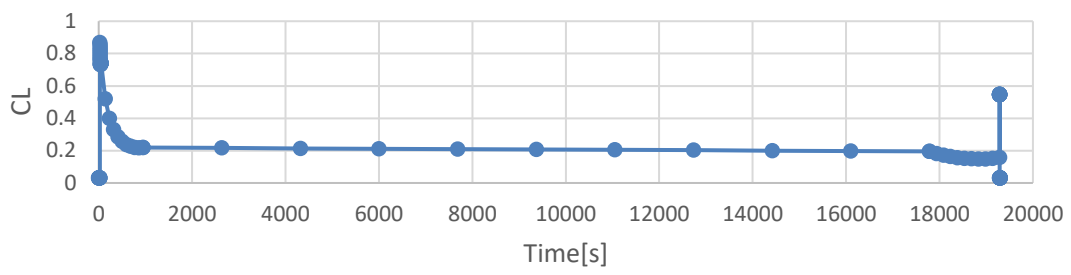
a. Altitude changes during the mission



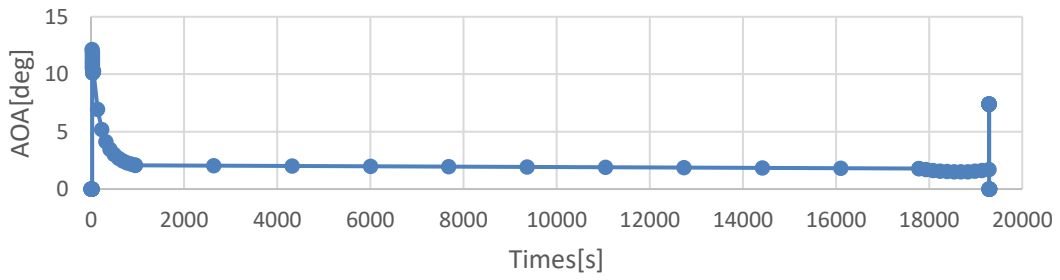
b. Mass changes during the mission



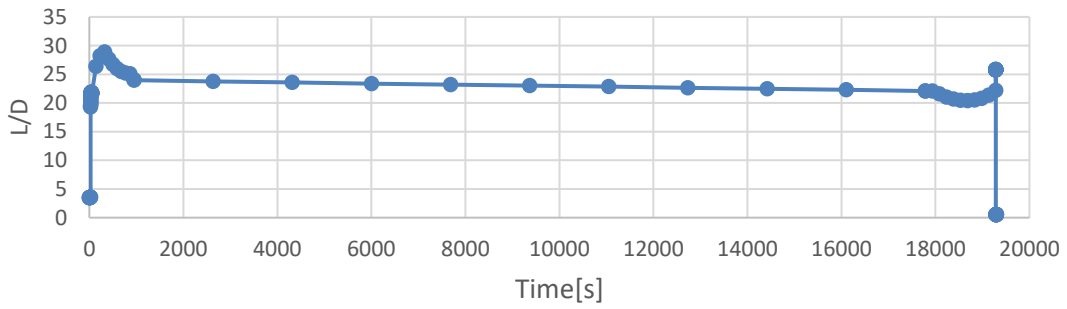
c. SFC changes during the mission



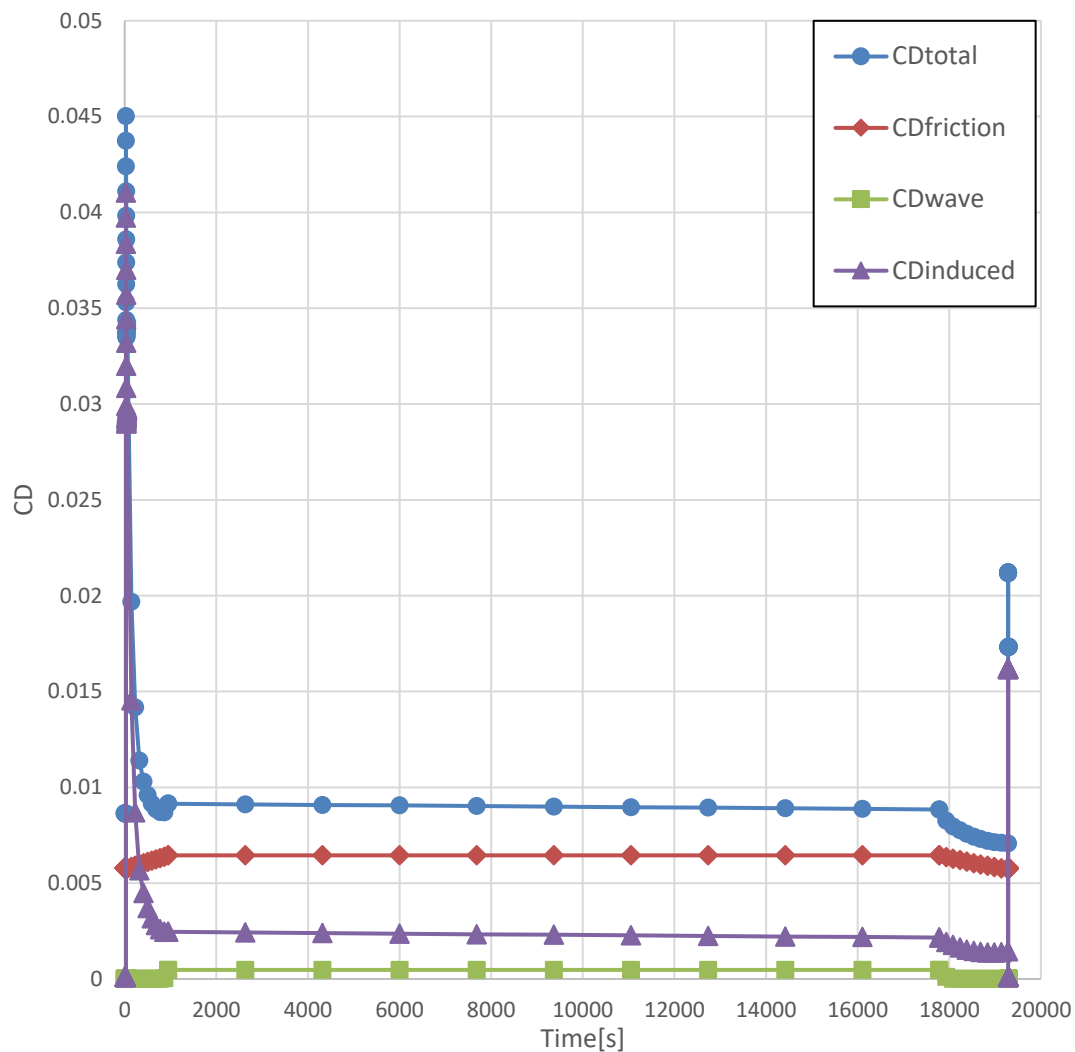
d. CL changes during the mission



e. AOA changes during the mission



f. L/D changes during the mission



g. CD changes during the mission

Figure 3-37 Mission performance for the SUGAR Ray

The mission's profile is depicted in Figure 3-37. The aeroplane climbs to 35000ft and maintains a consistent altitude while cruising. Additionally, Figure 3-37 subplot b shows that how the aircraft's weight lowers during the course of the trip as a result of the fuel burned. The particular amount of fuel consumed during the mission varies substantially. SFC values are low during the take-off and ascent segments due to the air's increased density and mass flow. The SFC is significantly higher during cruise, suggesting poor engine efficiency, yet the low cruise thrust required ensures a low fuel burn rate. The reduced cruise thrust requirement is necessary because the thrust must be equal to the vehicle drag

during cruise, as illustrated in Figure 3-37. Once the aeroplane begins to descend, the thrust significantly decreases while the drag increases.

The aerodynamic performance of the mission profile is depicted in Figure 3-37. The lift coefficients are presented in Figure 3-37 subplot d, which clearly demonstrates that the CL increases significantly during the take-off phase and then remains relatively constant during the cruise phase. During the take-off and landing portions, the angle of attack (as shown in Figure 3-37 subplot e.) is exceptionally high because this design is a clean configuration without any high-lift device and the mission solver is designed to meet the required lift of the aircraft by raising the angle of attack. This is also represented in the drag breakdown of the aircraft, as illustrated in Figure 3-37, where the induced drag makes the primary contribution to total drag throughout the ascent stage. As the mission progresses through the climb portions and approaches the cruise phase, friction drag becomes dominant. the aircraft's wave drag increases as well when it comes to cruise. It is necessary to explain that the angle of attack for the cruise phase is relatively high, because in this design, due to the lack of data, the NACA4 airfoils with 0 camber are selected, and the angle of attack and dihedral angle of all airfoil sections are set to 0 degrees, so the final geometry is not an optimal result. In the future, the angle of attack and dihedral angle of the airfoil sections as well as the camber of the outer wing section can be used as the optimisation variables to optimise the design to reduce the angle of attack in the cruise phase.

The CL and CD do not vary significantly during the cruise portion, as illustrated in Figure 3-37. Again, for the descending segment, the CL increases in proportion to the angle of attack. This is because the effect of spoilers is not currently simulated, and hence the solver increases the vehicle angle of attack to increase the induced drag and allows for a set rate of descent.

In general, the conceptual design methodology for the BWB, as implemented in GENUS, is capable of accurately simulating a BWB aircraft in a variety of aspects, including geometry, aerodynamics, mass estimation, and engine modelling, as well as providing remarkable detail for the various mission segments.

A detailed conceptual design process for a BWB configuration in GENUS has been presented in Appendix C.

4 Case Studies

4.1 Introduction

In this section, Comparison has been made in the four different aircraft concepts: conventional aircraft, the high-aspect-ratio-wing (HAW) aircraft, baseline BWB and BWB with BLI technology for different classes. These test cases are presented first, followed by their requirement analysis and design point analysis. After comparing the mass components of different concepts, the drag breakdowns are analysed for trends. At the end, the payload-range comparison are made to explore the advantages of different aircraft types.

4.2 Test Case Definition: Design Requirement

Given that the purpose of this study is to compare blended wing body aircraft to their conventional tube-and-wing equivalents, the mission criteria for BWB aircraft were chosen to be identical to those for tube-and-wing aircraft. Thus, the test cases are chosen to represent tube-and-wing aircraft in each of the size classes 200, 300, and 555 pax, allowing for a comprehensive investigation of the design space. The specifications for the tube-and-wing baseline aircrafts in three different classes are obtained from the Airbus A321, A330, and A380, respectively. Table 4-1 summarises the mission requirements and performance assumptions for each aircraft.

To better compare the performance characteristics of different concept aircraft, all aircrafts of different classes are designed to have the same fuel tank volume. tube-and-wing baseline configurations are modelled based on existing aircraft dimensions, and the HAW configurations are assumed to be not strut-supported and based on the baseline aircrafts with a proportional extension of the wing. The BWB configurations are completely tailless (no vertical stabilisers) and they are the optimisation results of the GENUS, in which the wingspan for airport limits and fuel tank volume are the constraints, and the mission fuel consumption is the optimisation objective.

In addition, boundary Layer ingestion technology is also considered in this work. By comparing the performance of the BWB aircraft with BLI and without BLI, the

benefit brought by BLI for different levels of BWB aircraft can be quantified and compared.

Table 4-1 Mission requirements for three categories

Aircraft Category	Narrow body	Wide body	Very Large Size
Seat Capacity	200	300	555
Design Range (km)	5950	12000	14000
Cruise Mach Number	0.78	0.80	0.85
Fuel Tank Volume (m ³)	30	100	270
Cruise Altitude (m)	11000		
Maximum manoeuvre normal load factor, n _z	2.5		
Passenger and baggage mass (kg)	83+32		

4.3 Test Cases

4.3.1 Narrow Body

As for narrow body aircrafts, A321 has been chosen as the conventional tube-and-wing configuration to be analysed in GENUS. Figure 4-1 shows the graphical representation of these three concepts, generated by the GENUS design environment. The basic performance for all concepts are shown in Table 4-2.

The mass breakdown of the four concepts, shown in Figure 4-2, demonstrates that the tube-and-wing aircraft has the lowest minimum take-off mass (MTOM) and the BWB has the highest one; to perform the same mission, the aircraft MTOM is raised from 82.1 t to 90.0 t, a 9.6% increase. The main factor for this difference is a significant, 21.1% increase in structure mass. Among the weight of the components, the wing weight and the fuselage weight have the biggest difference. BWB has the lowest wing weight due to shorter outer wing but the largest fuselage weight, which is more than twice the weight of the tube-and-wing aircraft fuselage, due to the entire pressurized cabin for BWB. As for this range level, the penalty from the fuselage exceeds the benefit from the outer wing, which result in the higher operational empty mass (OEM) for BWB. The larger

OEW of HAW aircraft is due to the increase in wing weight caused by the greater wingspan.

For the same mission, HAW and BWB aircraft have the lowest fuel consumption, 15.8 t, which is 6% lower than the fuel consumption of a tube-and-wing aircraft of 16.8 t. The cause of this reduction is due to the high aspect ratio of the HAW aircraft and large lifting surface of the BWB aircraft, which makes them both have a high lift to drag ratio. Figure 4-3 shows the comparison of drag build-up of these configurations. It can be seen that the CL required to sustain cruise (based on respective gross areas) is significantly smaller, 0.227 for the flying wing, compared to 0.617 for the conventional airliner and CD is larger for the conventional configuration, 0.0319 compared to 0.00967 for the BWB-200. BLI technology could accelerate the boundary layer on the surface of the aircraft, reducing the friction drag. With the application of BLI technology, the lift to drag ratio of BWB is increased by 2.38% and SFC is decreased by 1.1%, so that the fuel consumption is reduced by 4.04%. The less fuel consumption reduces the MTOM, which in turn reduces the OEM.

The payload-range of these four configurations is compared in Figure 4-4. The achievable range of HAW aircraft can be extended significantly compared to the similar conventional baseline and BWB configuration, which has the same range. BLI can improve the range of BWB to the same level with HAW aircraft.

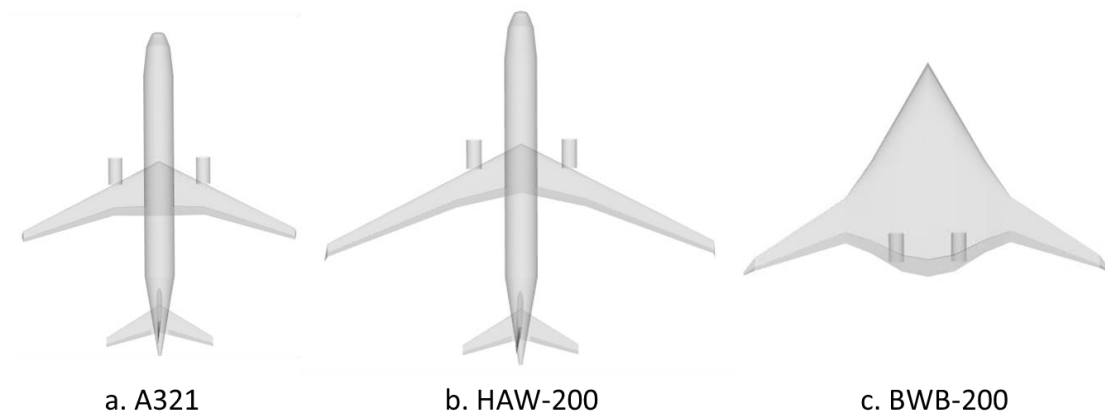


Figure 4-1 Graphical representation of narrow body aircrafts

Table 4-2 Cruise performance results for narrow body aircrafts

Parameter	A321-baseline	HAW-200	BWB-200	BWB-200+BLI
MTOM (kg)	82,071	88,573	89,956	89,284
OEM (kg)	42,250	49,756	51,166	51,133
Fuel Consumption (kg)	16,820	15,817	15,789	15,151
L/D	19.32	22.81	21.13	21.70
SFC(g/kN/s)	15.31	15.26	15.32	15.15
Total Surface Area (m ²)	775.75	837.55	828.82	828.82
Gross Wing Area (m ²)	133.71	157.35	397.68	397.68

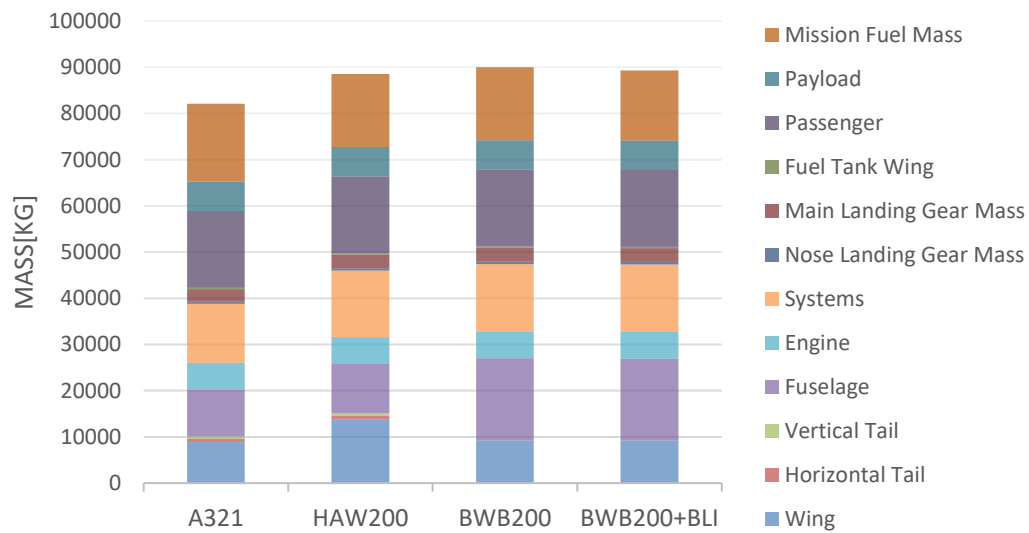


Figure 4-2 Mass breakdown for narrow body aircrafts



Figure 4-3 Drag build-up for narrow body aircrafts

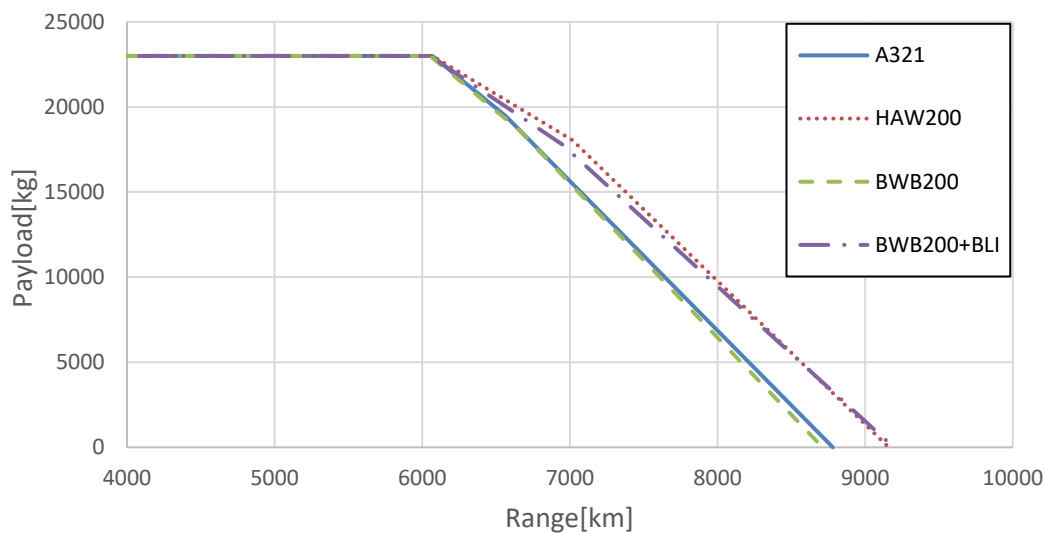


Figure 4-4 Payload-range diagram for narrow body aircrafts

4.3.2 Wide Body

A330-200 is chosen as the conventional tube-and-wing aircraft baseline model for the wide body class. Figure 4-5 shows the examples of tube-and-wing baseline, HAW and BWB concept generated from GENUS according to the mission requirement and Table 4-3 summarize the performances of these configurations.

The overall mass results are given in Figure 4-6. It shows that MTOM of BWB concept is obviously lower than its baseline and HAW counterpart due to its high L/D and light OEM.

The HAW aircraft has the highest lift-to-drag ratio of 21.85, however, the higher structural weight caused by the large aspect ratio of the wing makes its fuel consumption still lower than BWB concept.

The advantages of BWB aircraft are still reflected in the payload range of the aircraft, which is shown in Figure 4-7. Compared to the A330, the range of HAW and BWB has increased by 6.6% and 10.7% respectively. BLI technology can increase the revenue of BWB aircraft to 16.2%.

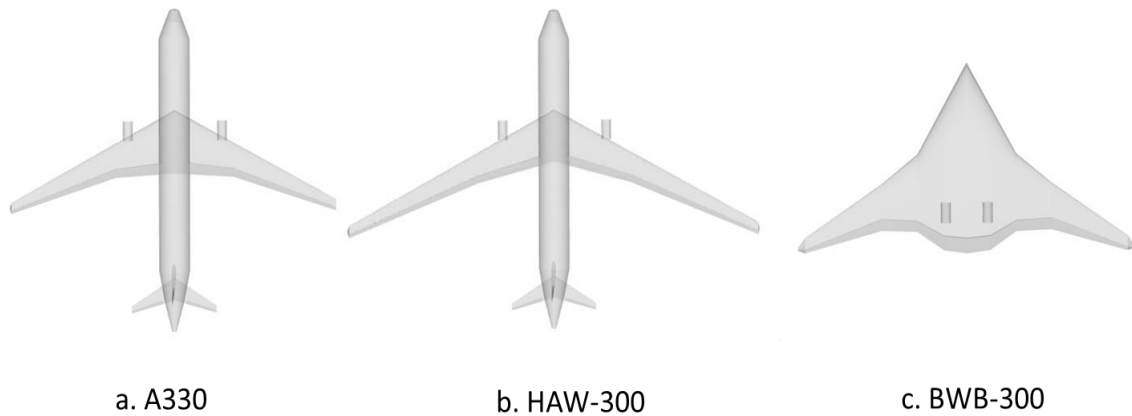


Figure 4-5 Graphical representation of wide body aircrafts

Table 4-3 Cruise performance results for wide body aircrafts

Parameter	A330-baseline	HAW-300	BWB-300	BWB-300+BLI
MTOM (kg)	204,336	214,271	191,305	187,097
OEM (kg)	101,989	117,390	96,689	95,854
Fuel Consumption (kg)	67,847	62,380	60,115	56,743
L/D	18.81	21.85	20.88	21.50
SFC (g/kN/s)	15.41	15.31	15.30	15.02
Total Surface Area (m ²)	1671.37	1800.73	1479.66	1479.66
Gross Wing Area (m ²)	357.14	412.59	709.2	709.2

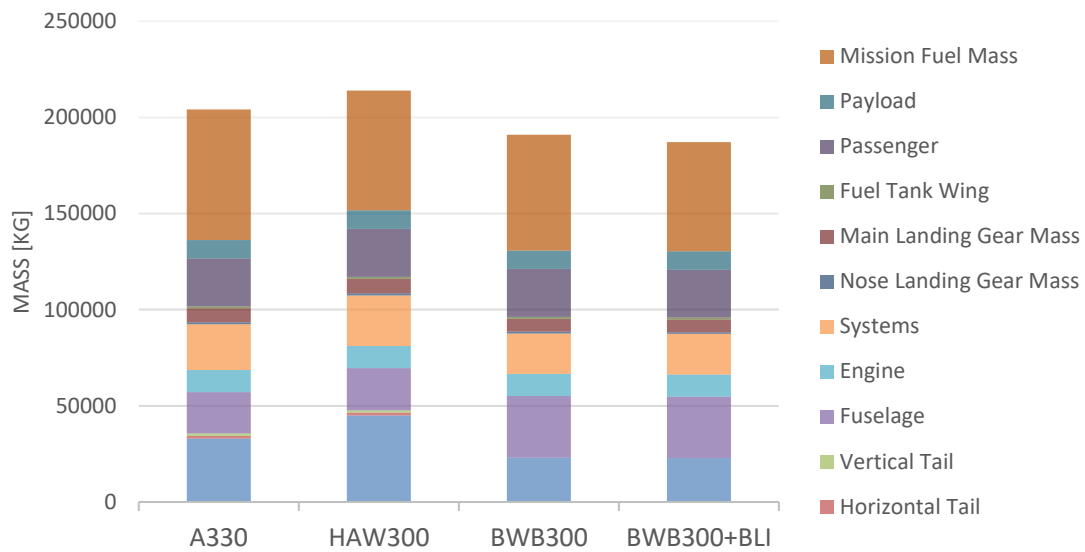


Figure 4-6 Mass breakdown for wide body aircrafts

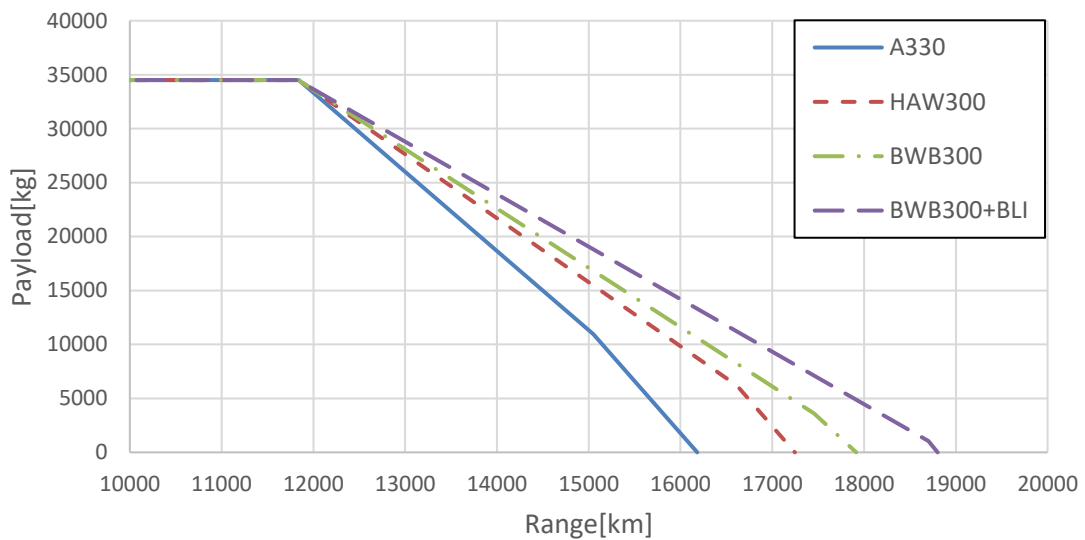


Figure 4-7 Payload-range diagram for wide body aircrafts

4.3.3 Very Large Size

A380 is analysed using the GENUS environment as the conventional tube-and-wing baseline aircraft, which is shown in Figure 4-8. Since the wingspan of A380 has reached the limit of airport width, there is no possibility to extend the

wingspan and increase the aspect ratio. Therefore, under the requirements of this task, there are only two configurations: tube-and-wing and BWB, and their mission performance is given in Table 4-4.

Similar to the long-range case, BWB has the lowest MTOM according to the mass result in Figure 4-9, which is 9.7% lighter than tube-and-wing aircraft. The reason for this difference is significant, 14.2% and 6.4% reduction in OEM and fuel mass respectively. The use of BLI increases the lift-to-drag ratio and reduces the SFC, which reduces the fuel consumption of BWB from 170t to 156t, a decrease of about 8%.

The payload range diagram are plotted in Figure 4-10. It can be seen that the achievable range of BWB increase by 15.5% compared to A380.

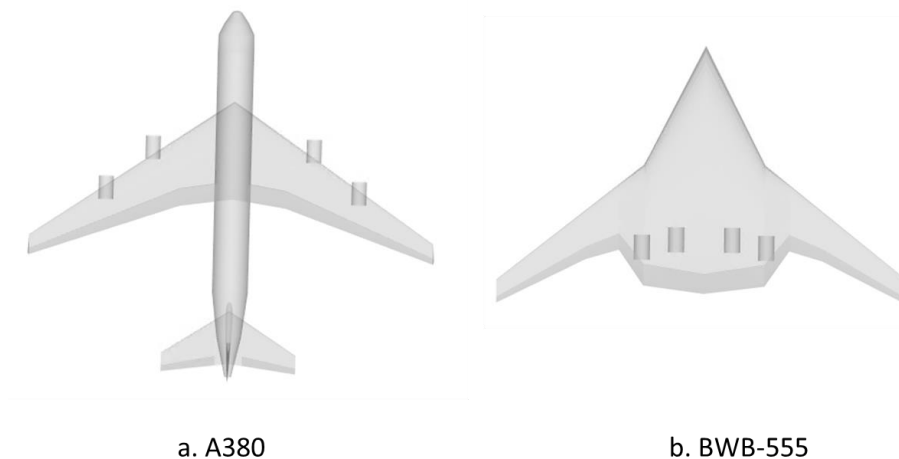


Figure 4-8 Graphical representation of very large size aircrafts

Table 4-4 Cruise performance results for very large size aircrafts

Parameter	A380-baseline	BWB-555	BWB-555+BLI
MTOM (kg)	512,083	462,549	449,674
OEM (kg)	266,929	228,929	227,803
Fuel Consumption (kg)	181,328	169,795	156,046
L/D	18.25	21.10	21.68
SFC (g/kN/s)	15.37	15.33	14.97
Total Surface Area (m ²)	3389.66	2880.66	2880.66
Gross Wing Area (m ²)	858.43	1370.02	1370.02

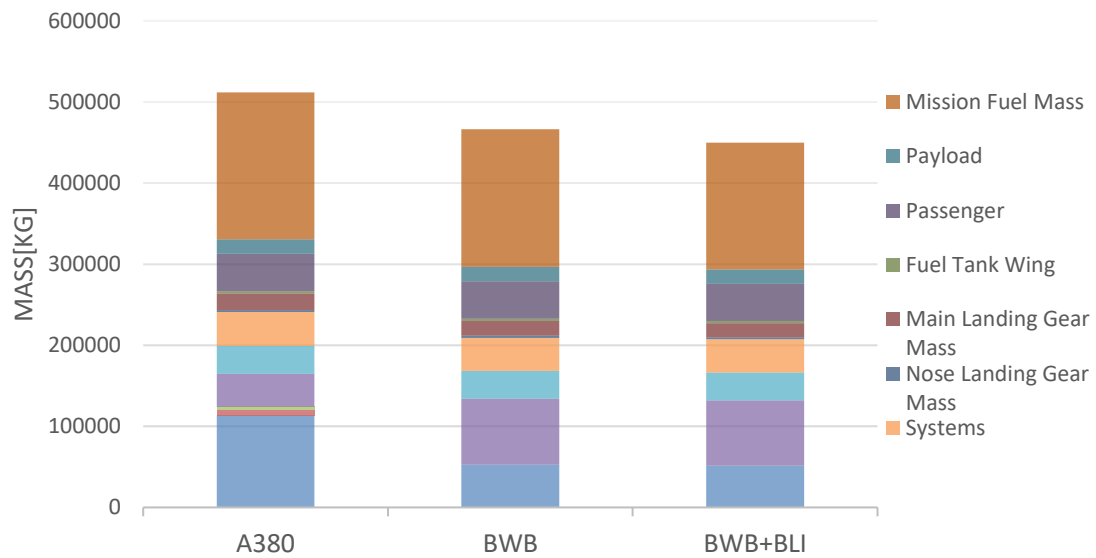


Figure 4-9 Mass breakdown for very large size aircrafts

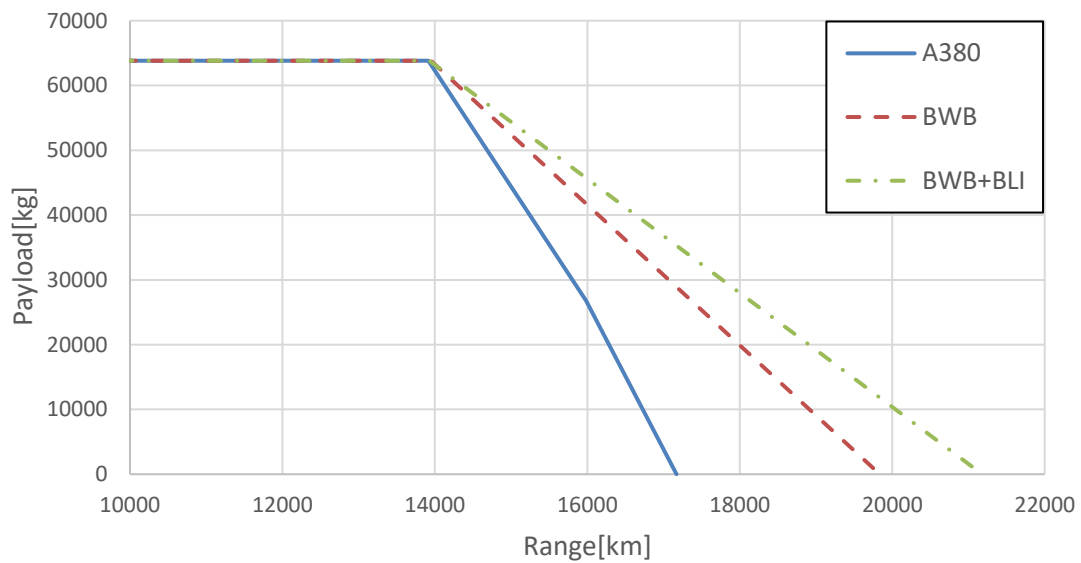


Figure 4-10 Payload-range diagram for very large size aircrafts

4.4 Results and Discussion

With the same engines and the same mission requirements, there are two main factors that have a major impact on the fuel consumption of the aircraft: structure weight and lift-drag ratio. Therefore, the final fuel consumption is a trade-off

process. The comparison of key parameters for different type of aircrafts are given in Table 4-5.

Table 4-5 Comparison of relative changes in key aircraft parameters with respect to tube-and-wing aircraft

Parameters (%)	Narrow Body				Wide Body				Very Large Size		
	A321	HAW -200	BWB -200	BWB- 200+BLI	A330	HAW -300	BWB -300	BWB- 300+BLI	A380	BWB -555	BWB- 555+BLI
MTOM	-	7.9	9.6	8.79	-	4.9	-6.4	-8.4	-	-9.7	-13.0
OEM	-	17.8	21.1	21.0	-	15.1	-5.2	-6.0	-	-14.2	-15.4
Fuel consumption	-	-6.0	-6.1	-10.0	-	-8.1	-8.4	-13.5	-	-10.4	-16.9
L/D at cruise	-	18.1	9.4	12.3	-	16.2	11.0	14.3	-	15.6	19.5

The comparison results demonstrate that the gains in L/D grow with aircraft size, indicating that the BWB is more favourable in terms of drag reduction at larger aircraft sizes. These characteristics bolster the case for BWB aircraft being more aerodynamically efficient than tube-and-wing aircraft. This finding is supported by Reist and Zingg [138], who discovered comparable results.

In terms of structure mass, trends in the mass reductions favour larger aircraft where the best results are achieved in the 555-passenger class. At the small size (200 pax class), the BWB's structure weighs 21% more than a conventional aircraft due to the largest fuselage weight. As the size of the aircraft increased, the structural weight advantage of the BWB began to emerge, from -5.2% to -14.2%.

The application of BLI technology further expands the advantages of BWB, and the gains from BLI increase with aircraft size, such as from 2.9% to 3.9% increase in lift to drag ratio and from 3.9% to 6.5% reduction in fuel consumption compared to the initial BWB airplanes. This is because with the increase of aircraft size, the length of the aircraft central body will also increase, and the longer path of air flowing through the fuselage into the engine will increase the thickness of the

boundary layer into the engine. The more the ingested boundary layer, not only the friction drag will be reduced, but also the SFC of the engine will be reduced, thus reducing the total fuel consumption.

In conclusion, the preliminary comparisons of BWB with tube-and-wing aircraft demonstrate that the BWB is more aerodynamically and structurally efficient, particularly for larger aircraft. The mass estimates predict that the operational empty weight of a BWB will be greater than that of a tube-and-wing aircraft at modest sizes (200 pax class) and will be lower as the passenger count exceeds 300. When combined with the increased aerodynamic efficiency, this results in a reduction in fuel consumption of up to 10.4% for a very long-range 555-passenger aircraft. Boundary layer ingestion technology is one of the key technologies of BWB aircraft, which could increase the profit of BWB aircraft. Given the current state of BWB implementation and the degree of connection between design disciplines, it is reasonable to predict that the quantitative conclusions of this study comparing BWB and tube-and-wing aircraft will alter as BWB implementation progresses. Qualitative results and trends, on the other hand, should remain valid.

5 Design and Analysis of BWB Aircraft

5.1 Introduction

The literature review demonstrates the strong interdisciplinary nature of the BWB aircraft design process; hence, this section will investigate and analyse the implications of several aspects affecting the BWB's design. The outcome of this study can give a design with the necessary facts to make educated design decisions during the BWB synthesis.

In this section, the design parameters are divided into two parts: Geometry parameters including central body sweep and outer wing sweep, and operational parameters including cruise altitude and cruise Mach number. The effects of these parameters to the mission performance for a subsonic, tailless BWB configuration are investigated.

5.2 Mission and Baseline Design

The mission requirements for the baseline BWB concept are specified in Table 5-1, which is similar with A380 class 'Very Long Range' standards.

Table 5-1 Mission requirements for the baseline BWB concept

Requirement	Value
Seat Capacity	9+80+466, 3 classes configuration
Design Range (km)	14000
Cruise Mach Number	0.80
Cruise Altitude (m)	10059
Maximum manoeuvre normal load factor	2.5
Passenger and baggage mass (kg)	83+32

According to the mission requirements, a baseline configuration with NACA4 aerofoils of varying thicknesses has been designed in the GENUS. The major components have been sized and located through the mass estimation module

and the clashes check have also been carried out and included in the packaging routines. It is worth noting that four Trent-900 engines with sea level maximum thrust of 366 kN and cruise SFC of 15.7 g/(kN·s) has been arranged on the design, and for all trade studies, the propulsion system keeps constant. A three-view schematic of the packaged design is shown in Figure 5-1. The specific geometry parameters and performance results from the analysis of GENUS are list in Table 5-2 and Table 5-3 respectively.

Table 5-2 Geometry parameters for baseline BWB configuration

Parameter	Value
Central body sweep (deg)	63
Outer sweep	38.3
Chord0 (m)	45
Chord1(m)	22
Chord2 (m)	12
Chord3 (m)	9
Chord4 (m)	4.23
Span0 (m)	11
Span1 (m)	4.5
Span2 (m)	6.0
Span3 (m)	16.25

Table 5-3 Geometry and performance results for baseline BWB from GENUS

Parameter	Value
Total span (m)	75.44
Reference area (m ²)	1230.52
AR	4.62
MTOM (kg)	432711.8
OEM (kg)	208954.4
Fuel consumption (kg)	161191.4
L/D at cruise	21.06

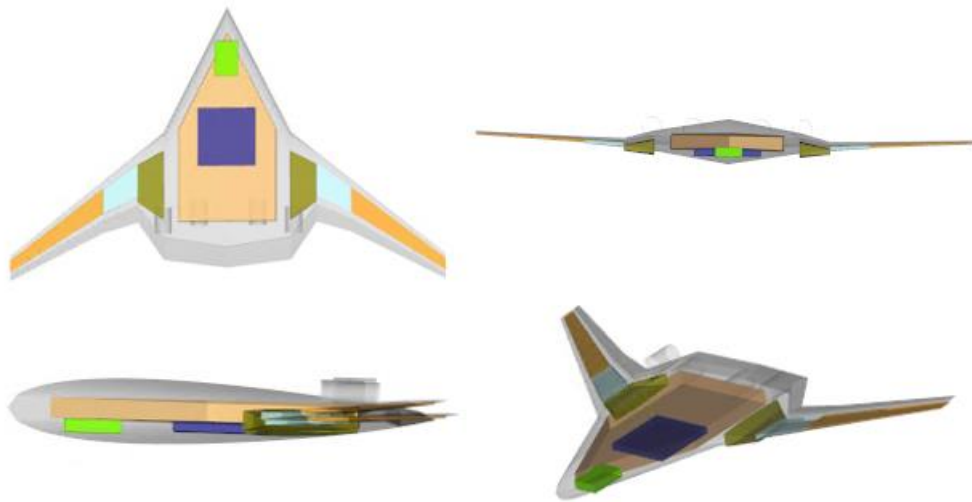
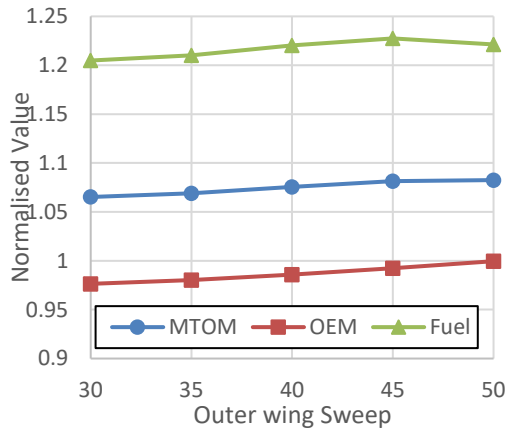


Figure 5-1 three view schematic of the baseline BWB configuration

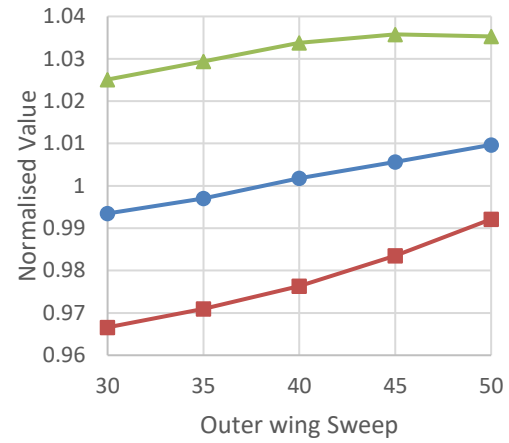
5.3 Parameter Sensitivity Analysis

5.3.1 Geometry Parameters

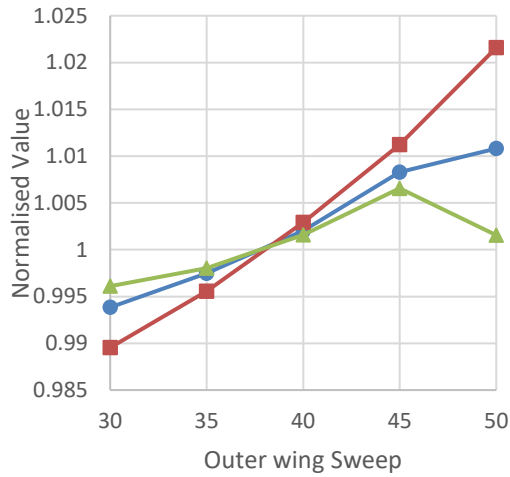
The baseline mission requirements have been evaluated for configuration of varying both central body and outer wing leading edge sweep angles under constant wing area. Figure 5-2 shows the MTOM, OEM and fuel mass normalised with respect to the baseline configuration. It can be seen that the MTOM and OEM increase as the outer wing sweep increases for a fixed central body sweep. The total fuel consumption shows different trend for different central body sweep. For a low central body sweep of 40° and 50° , the fuel consumption keeps constant with the increasing outer wing sweep. However, for the central body sweep of 63° , the fuel mass displays the tendency rising up at the beginning and declining for outer wing sweep above 45° , while it keeps falling as the outer wing sweep increases.



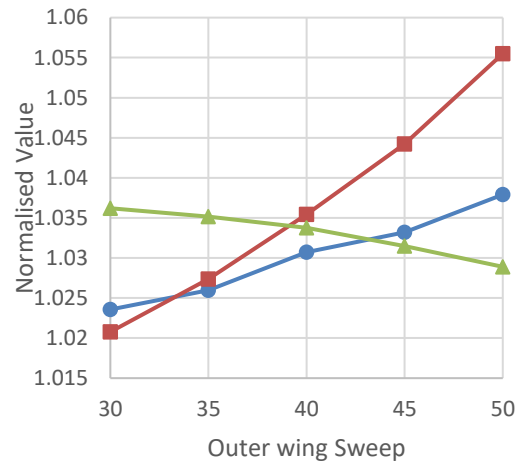
a. Central body sweep 40 at Mach 0.8



b. Central body sweep 50 at Mach 0.8



c. Central body sweep 63 at Mach 0.8



d. Central body sweep 70 at Mach 0.8

Figure 5-2 MTOM, OEM and fuel mass for different central body sweep

The reason for the different trend in the fuel consumption mass is showed below. The fuel consumption calculation is given by Eq. (5-1).

$$\begin{aligned}
 Fuel &= time * SFC * thrus = \frac{range}{Mach * a} * SFC * \frac{mass}{L/D} \\
 &= \frac{range}{Mach * a} * SFC * \frac{Fuel + OEM}{L/D}
 \end{aligned} \tag{5-1}$$

Set $A = \frac{range}{Mach * a} * SFC$, Mach is fixed, SFC is related to the height and Mach number, so A is fixed for a given mission profile and the Eq. (5-1) can be transferred to Eq. (5-2), which shows there is a positive correlation between fuel consumption mass and factor $\frac{OEM}{L/D}$.

$$Fuel = \frac{A * OEM}{L/D - A} \quad (5-2)$$

The lift to drag ratio (L/D) vs different sweep leading edge at Mach 0.8 is plotted in Figure 5-3. For a constant central body sweep, the L/D increase slightly as the outer wing sweep increases. As for the different central body sweep angles, there is sharp rise from 40° to 63° and slight fall from 63° to 70°. This behaviour is clearly seen in the drag build-up results shown in Figure 5-4, which reveals that central body sweep has the greatest influence on the wave drag, and the larger the sweep angle, the smaller the wave drag and for the sweep angle of the outer wing, when central body sweep is constant, the wave drag reduction is small from 30° to 45°, and large from 45° to 50°. In addition, it worth noting that although wave drag decreases as sweep angle increases from 63° to 70°, induced drag also increases, which leads to a decrease in the total lift-to-drag ratio.

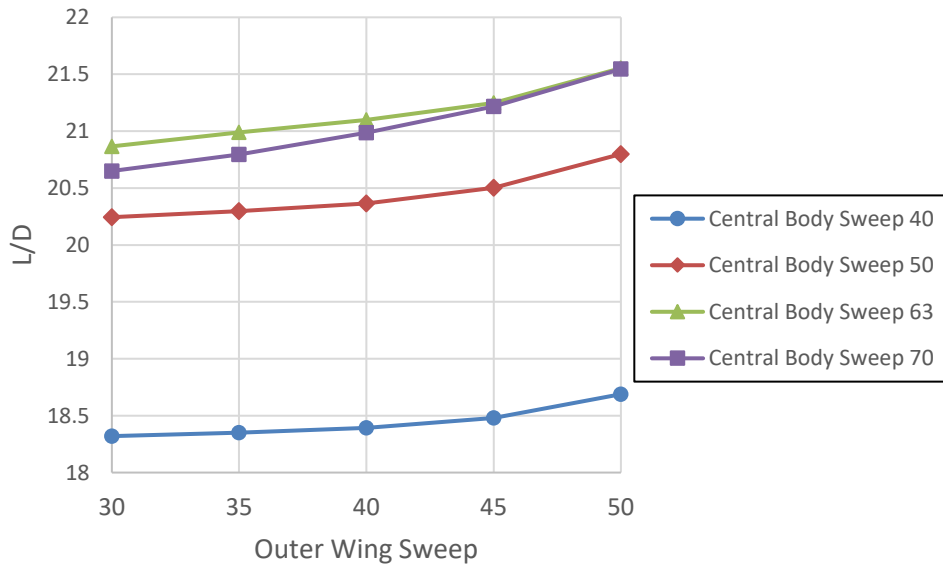
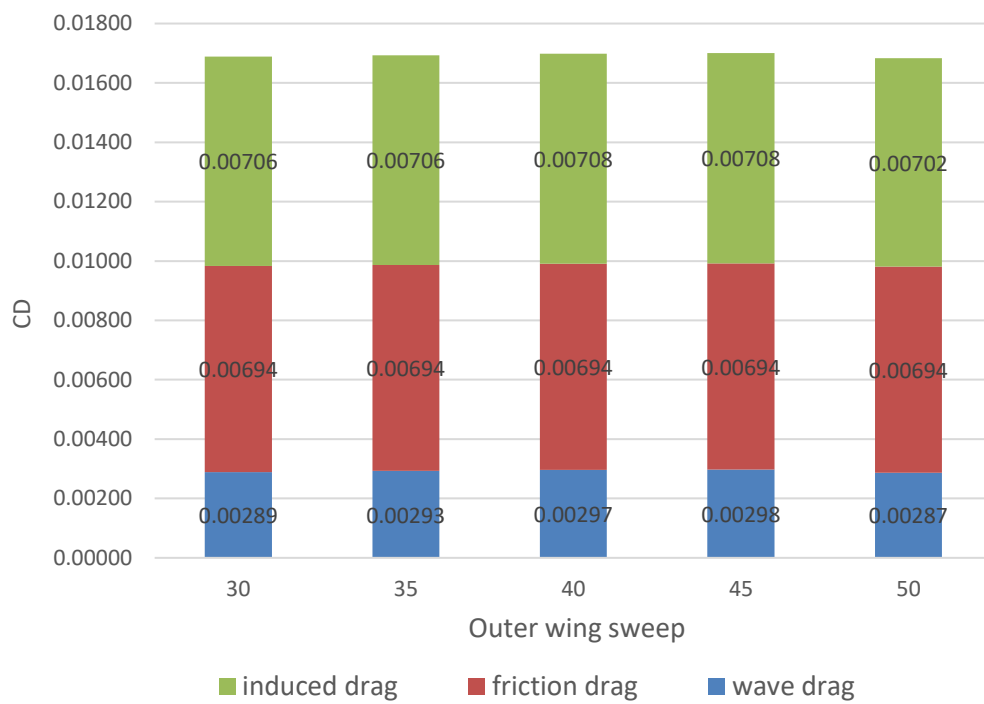
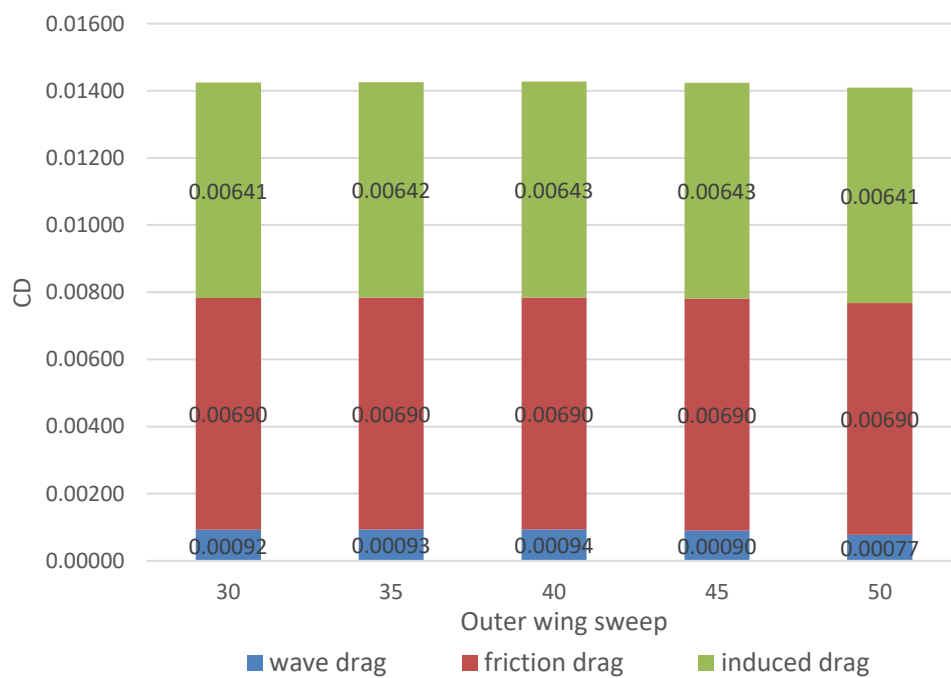


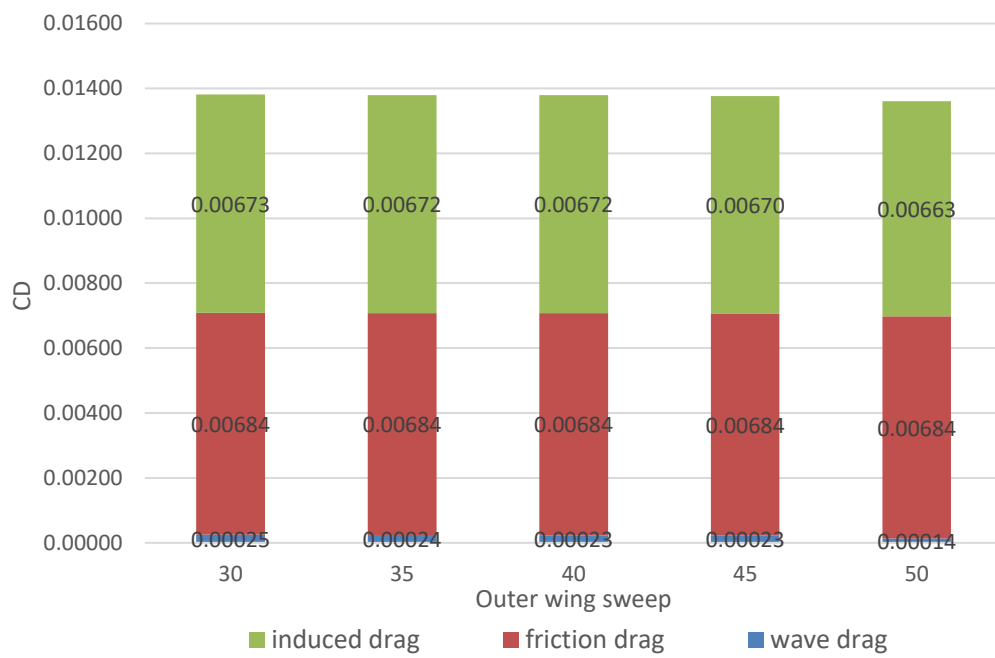
Figure 5-3 Lift to drag ratio for different sweep at Mach 0.80



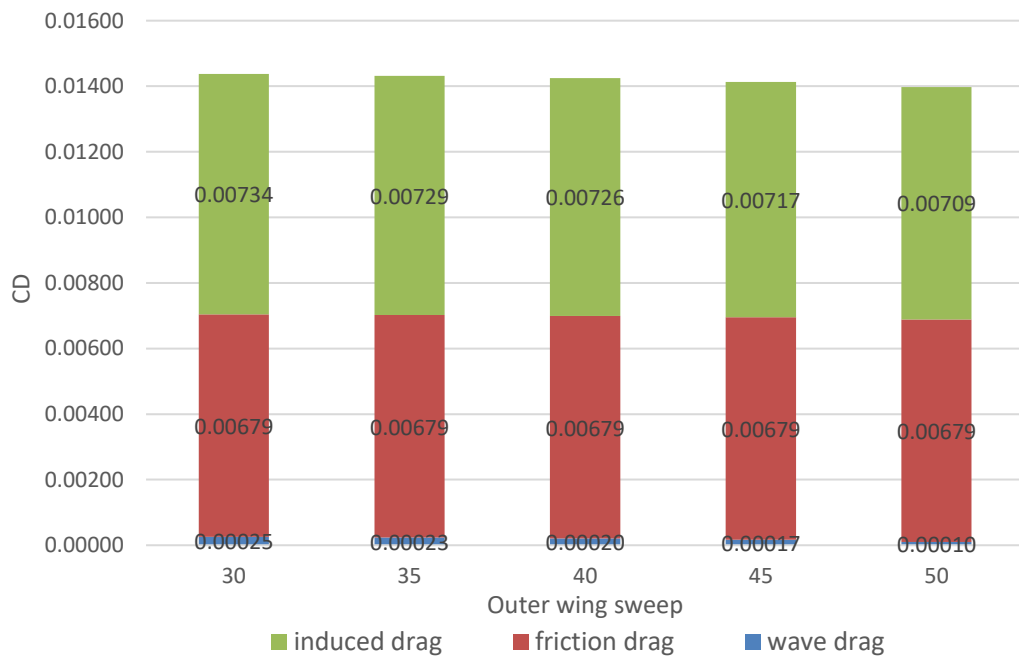
a. Central body sweep 40 drag build-ups



b. Central body sweep 50 drag build-ups



c. Central body sweep 63 drag build-ups



d. Central body sweep 70 drag build-ups

Figure 5-4 Drag build-ups for different central body sweep

The OEM result for different leading-edge sweep shown in Figure 5-5 shows that increased sweep angle increases the structural weight for the BWB configuration, which result in the rise in the fuel consumption.

From the analysis of the L/D and OEM result, it can be seen that the larger sweep angle increases the weight and fuel consumption of the structure, but also increases the lift-to-drag ratio and thus reduces the fuel consumption. Therefore, the final fuel consumption is a trade-off process, and the central body sweep angle plays a decisive role. The factor $\frac{OEM}{L/D}$ is plotted in Figure 5-6, which shows the similar trend as fuel consumption result from performance module in GENUS, shown as Figure 5-7. The main difference between the two results is fuel consumption for central body sweep of 50° and 63°. That is because the L/D keeps constant in the calculation of the factor $\frac{OEM}{L/D}$ and it changes with the updated aircraft mass in the performance analysis of GENUS. However, the same conclusion can be obtained from these two results is that the BWB aircraft with central body sweep of 63° has the lowest fuel consumption.

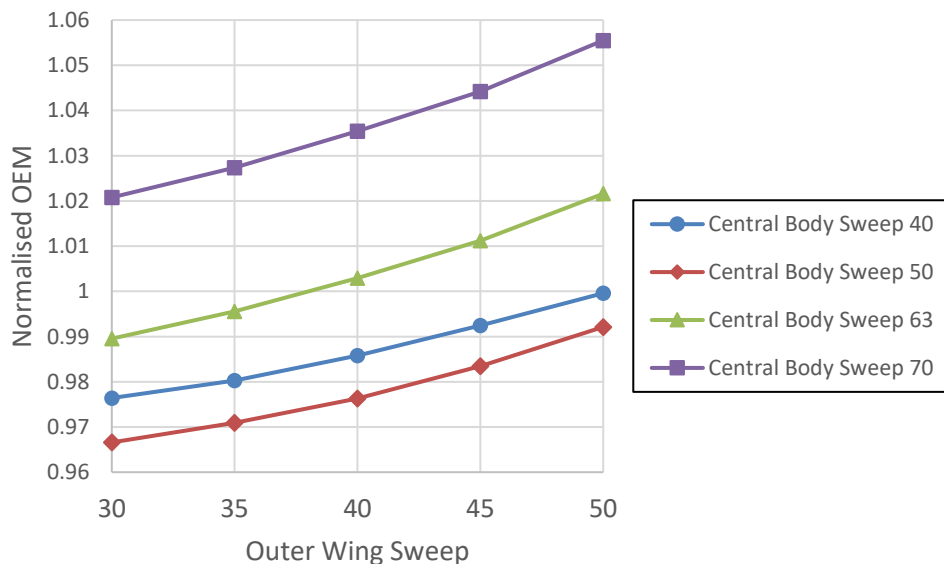


Figure 5-5 OEM for different sweep at Mach 0.80

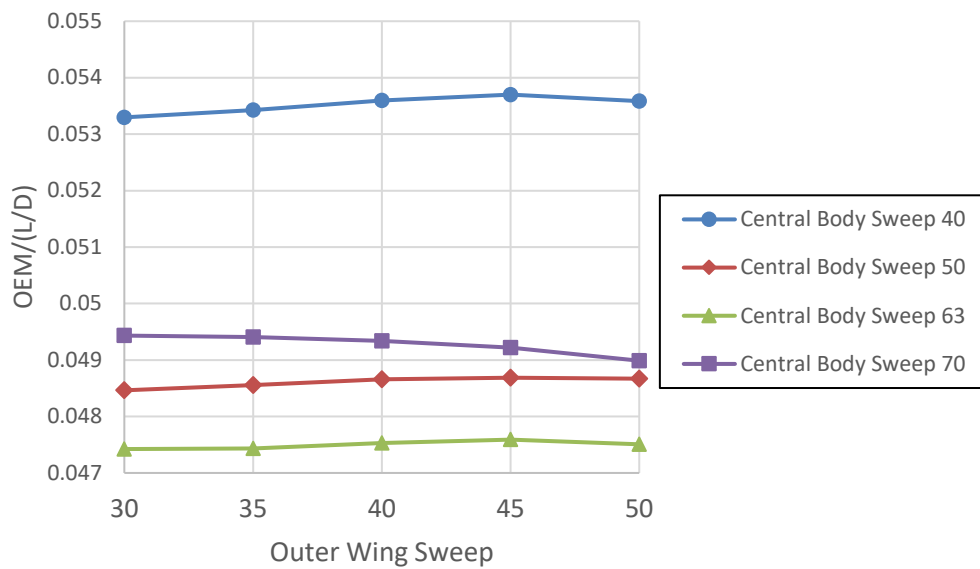


Figure 5-6 Fuel consumption factor

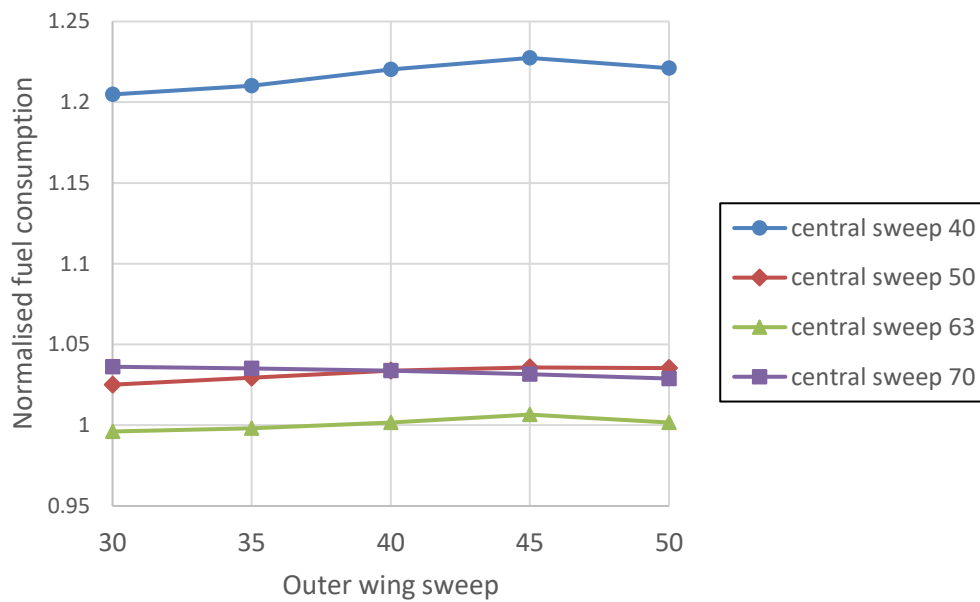


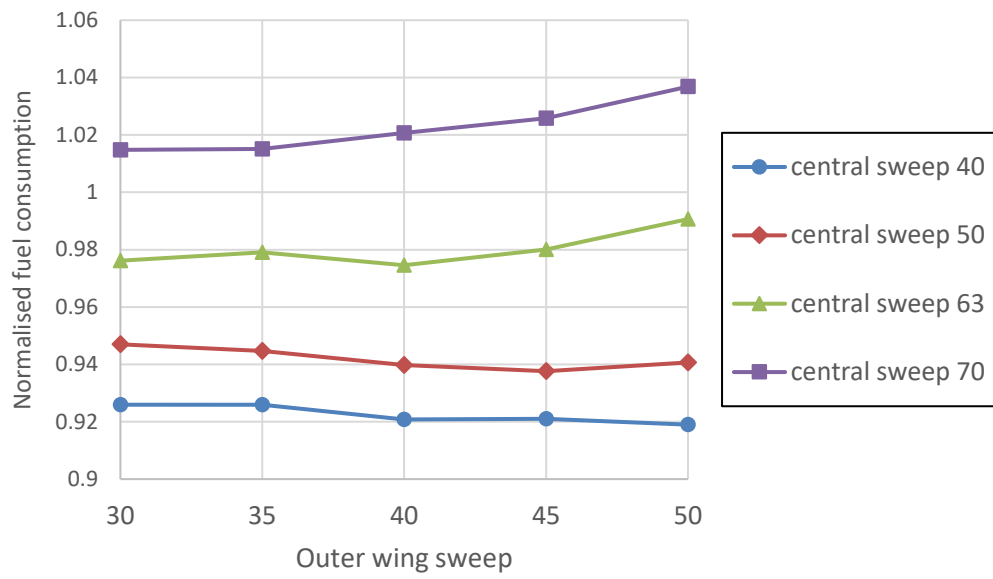
Figure 5-7 Fuel consumption for different sweep at Mach 0.80

Figure 5-8 shows the fuel consumption for different sweep at different Mach number. It is worth noting that, in the case of high Mach number, low central body sweep angle will lead to large wave drag, resulting in low lift-to-drag ratio and unable to obtain a convergence solution from GENUS. Therefore, when Mach number is 0.85, the central body sweep angle starts from 50°, and when Mach number is 0.9, the central body sweep angle starts from 63°.

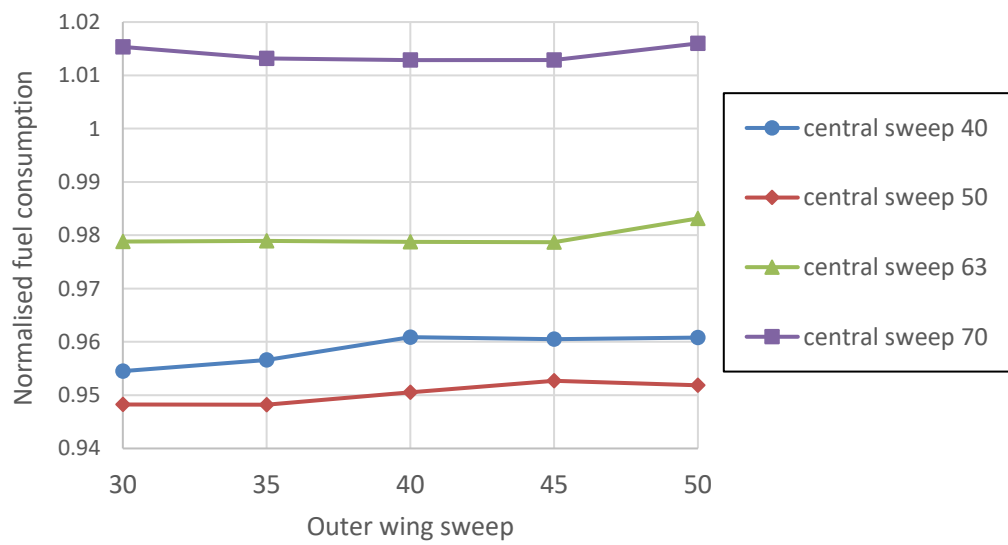
From Figure 5-8, it can be seen that the fuel consumption shows high dependence on the central body sweep angle which presents a layered distribution. When the Mach number is from 0.7 to 0.85, outer wing sweep has small effect on the fuel consumption, and only when the Mach number is 0.9, the fuel consumption decreases significantly with the increase of the outer wing sweep angle.

In addition, for different Mach number, there is an optimal central body sweep angle to realise the minimum fuel consumption. With the increase of Mach number, the optimal sweep also increase. This is can be explained by the drag divergence Mach number as a function of central body sweep angle shown in Figure 5-9. However, it ought to be noted that increase in central body sweep angles is accompanied with corresponding extension in the length of the central body which can increase the structure weight, in order to accommodate the given number of passengers. Consequently, at low Mach number, the small sweep angle makes the fuel consumption lower, because the wave drag of low Mach number has less influence, and the structure weight has more influence. With the increase of Mach number, the effect of wave drag becomes larger.

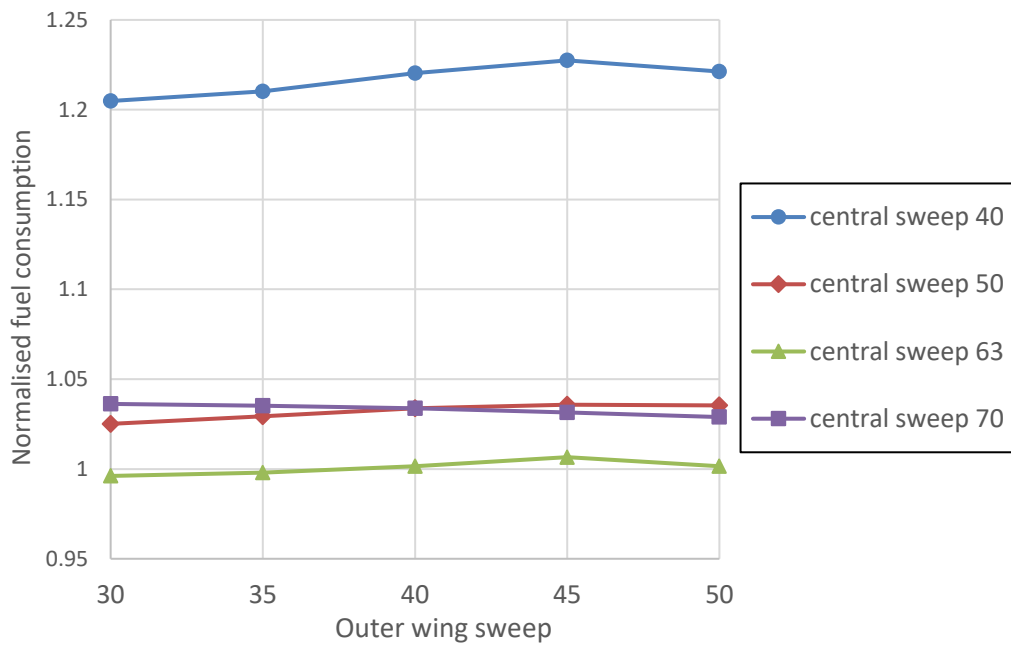
For a fixed outer wing sweep 40° it can be known that there is a fuel-efficient central body sweep angle for a specific cruise Mach number, which is shown in Figure 5-10.



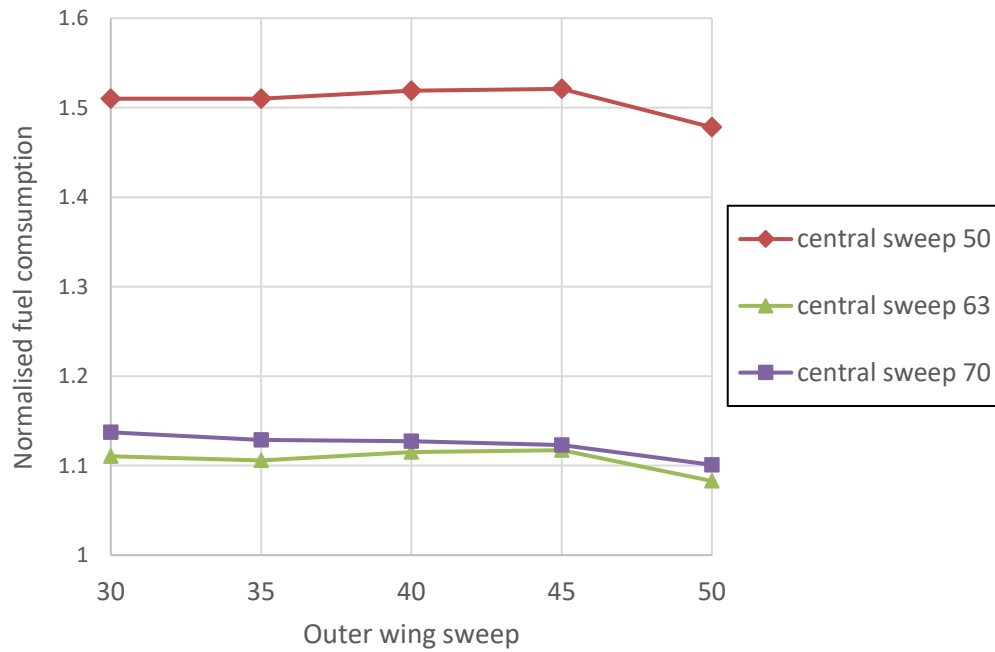
a. Fuel consumption at Mach 0.70



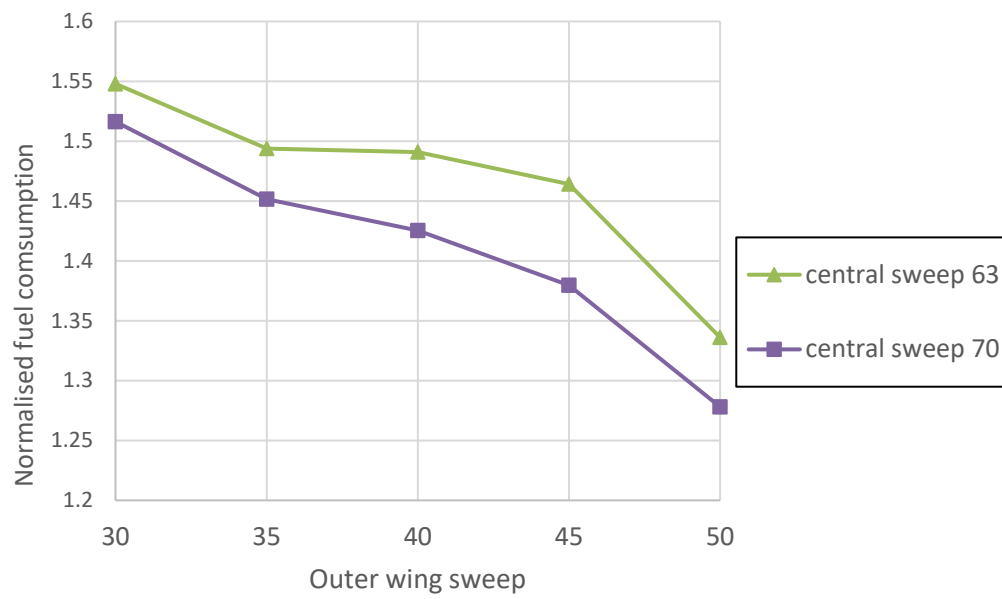
b. Fuel consumption at Mach 0.75



c. Fuel consumption at Mach 0.80



d. Fuel consumption at Mach 0.85



e. Fuel consumption at Mach 0.90

Figure 5-8 Fuel consumption for different sweep at different Mach number

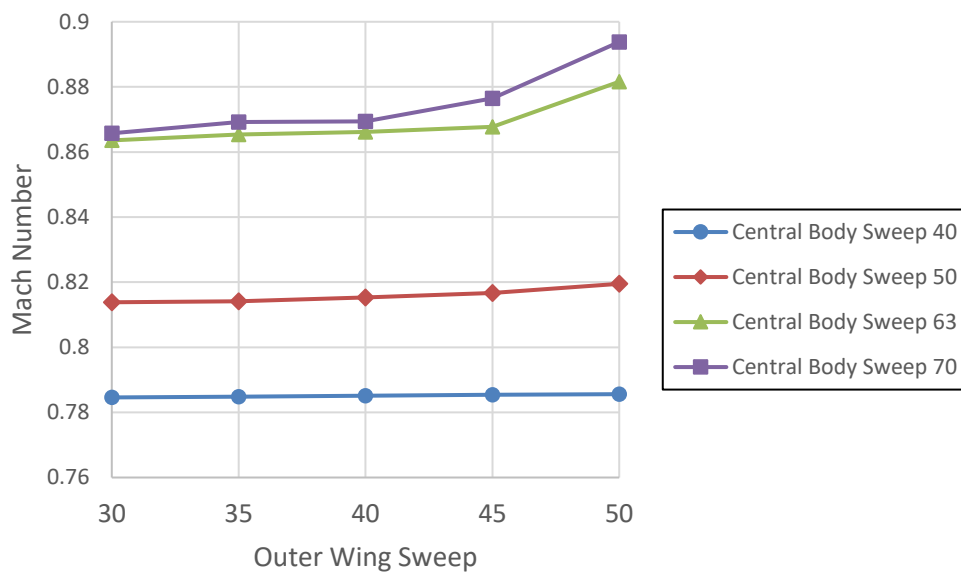


Figure 5-9 Drag divergence Mach number for different sweep

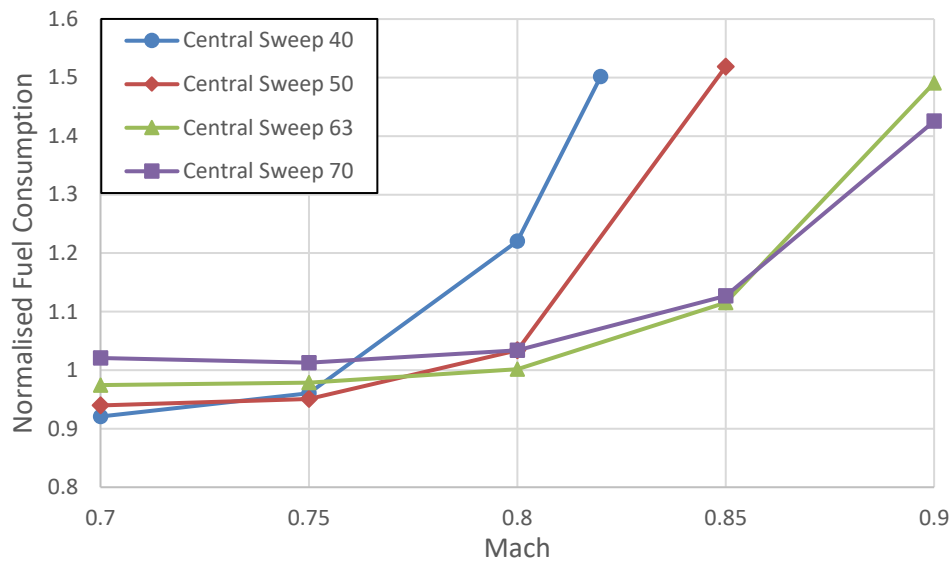
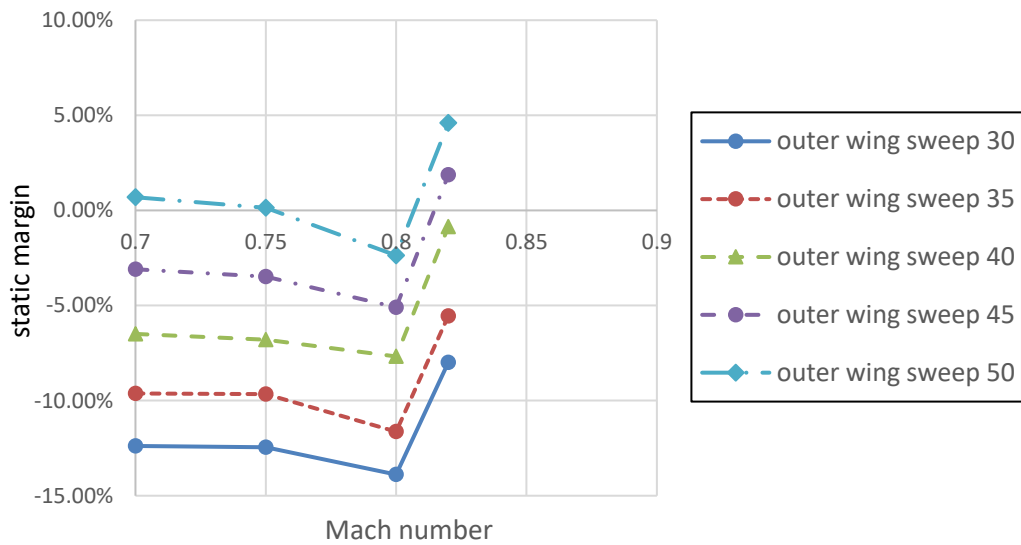
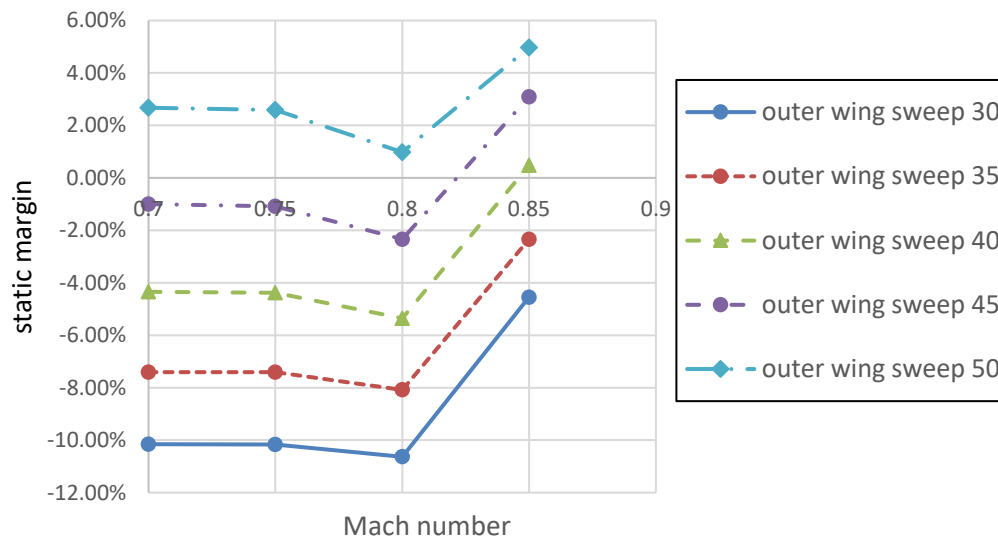


Figure 5-10 Fuel consumption for different Mach number and central body sweep (fixed outer wing sweep 40°)

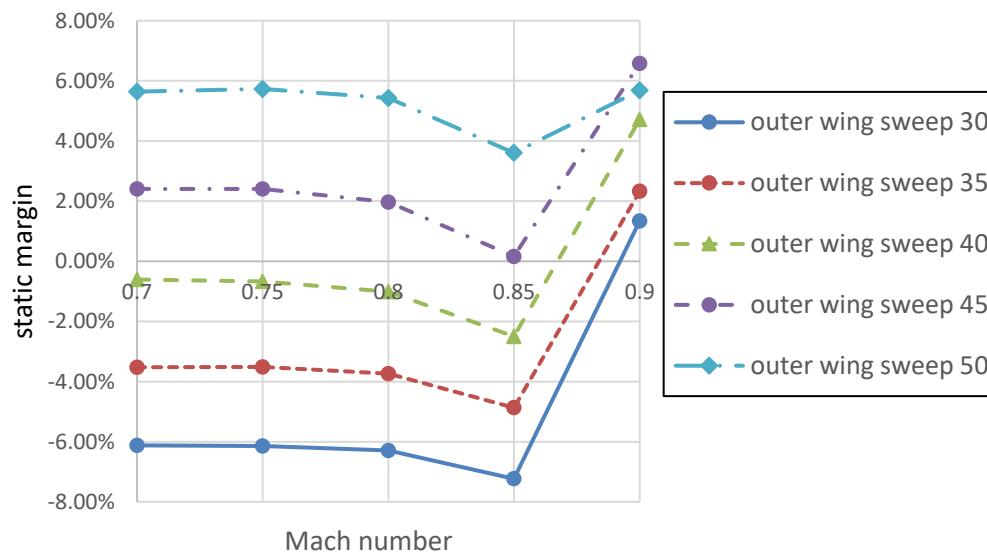
Following that, the influence of Mach number and sweep on the static margin was also determined for the test geometry. As illustrated in Figure 5-11, the result demonstrates an increase in static margin as the sweep angle increases. This is because increasing the sweep angle causes a rearward shift in the aircraft's aerodynamic centre, hence enhancing or increasing the static margin.



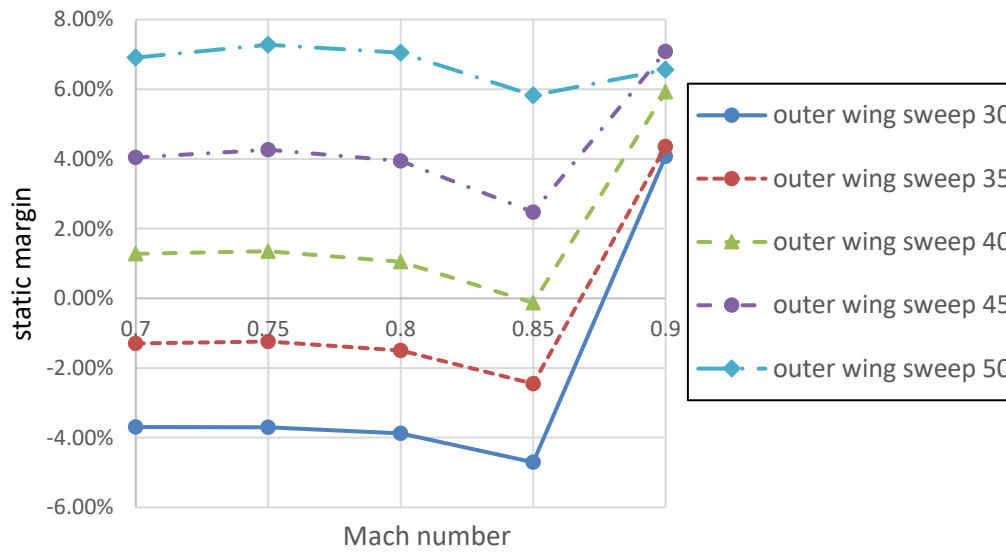
a. Central body sweep 40 static margin



b. Central body sweep 50 static margin



c. Central body sweep 63 static margin



d. Central body sweep 70 static margin

Figure 5-11 Static margin for different sweep

5.3.2 Operational Parameters

This section investigates the fuel consumption of the baseline BWB aircraft at various subsonic Mach number and flight altitudes.

The overall fuel consumption at various Mach numbers and altitudes is depicted in Figure 5-12. For every cruise Mach number, there is a range of altitudes where the fuel consumption is the lowest, which estimates the fuel-efficient altitude for a specific operational Mach number.

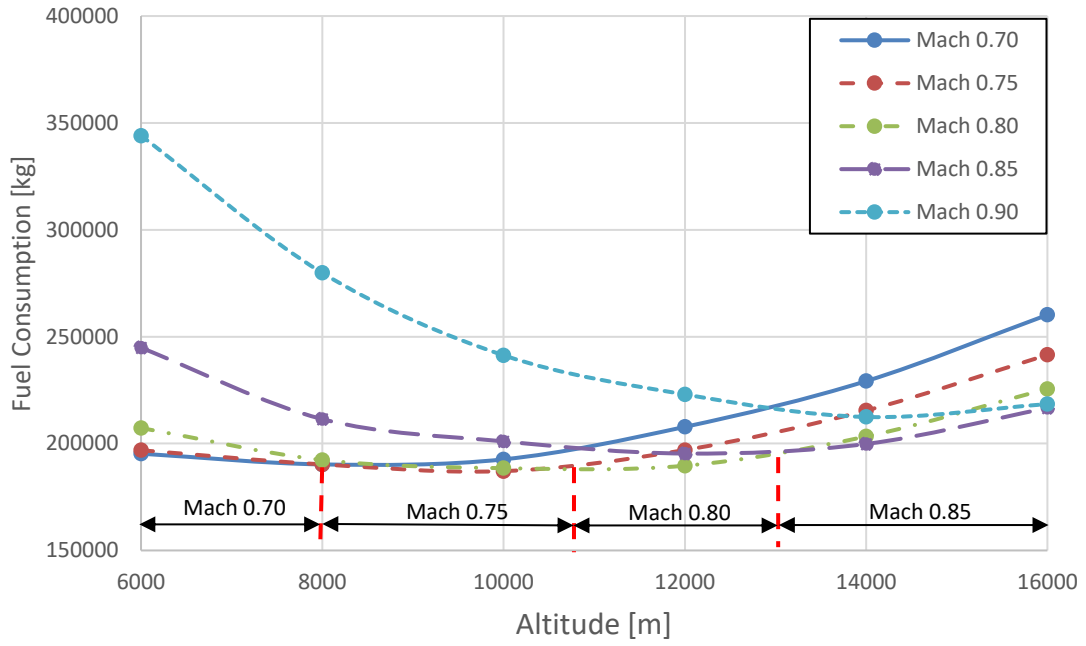


Figure 5-12 Fuel consumption for various Mach numbers and altitudes

The following provides an explanation for the results in Figure 5-12. The fuel consumption calculation is given by Eq. (5-3). For a given subsonic cruise range, the start mass is also the same, therefore, the total fuel is proportional to the factor $\frac{SFC}{L/D * Mach * a}$

$$Fuel = time * SFC * thrust = \frac{range}{Mach * a} * SFC * \frac{mass}{L/D} \propto \frac{SFC}{L/D * Mach * a} \quad (5-3)$$

The speed of sound in different altitudes in this research (6000 to 16000m) is plotted in Figure 5-13. L/D and SFC at subsonic speeds are shown in Figure 5-14 and Figure 5-15 respectively. Then the fuel consumption factor $\frac{SFC}{L/D * Mach * a}$ is shown in Figure 5-16, revealing the same results as Figure 5-12.

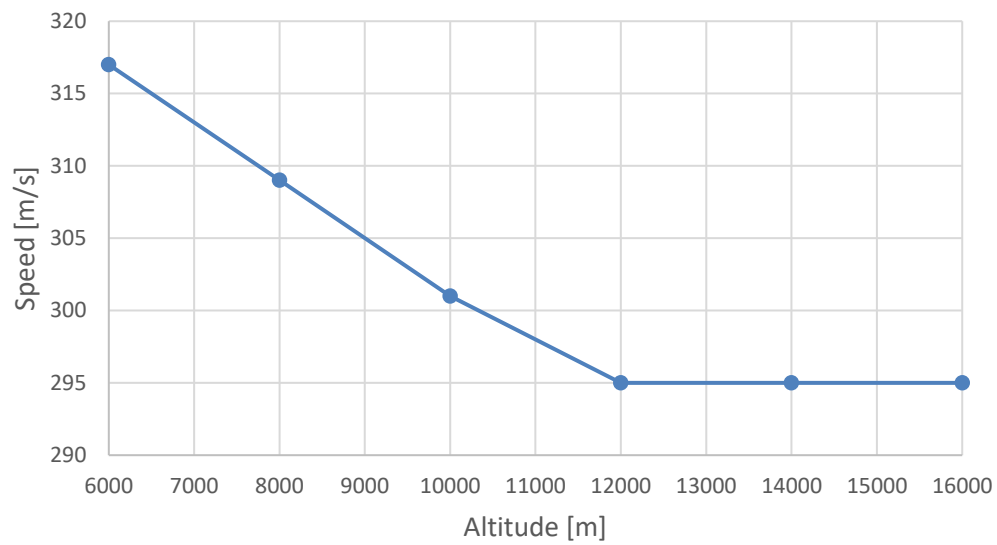


Figure 5-13 Speed of sound at different altitude

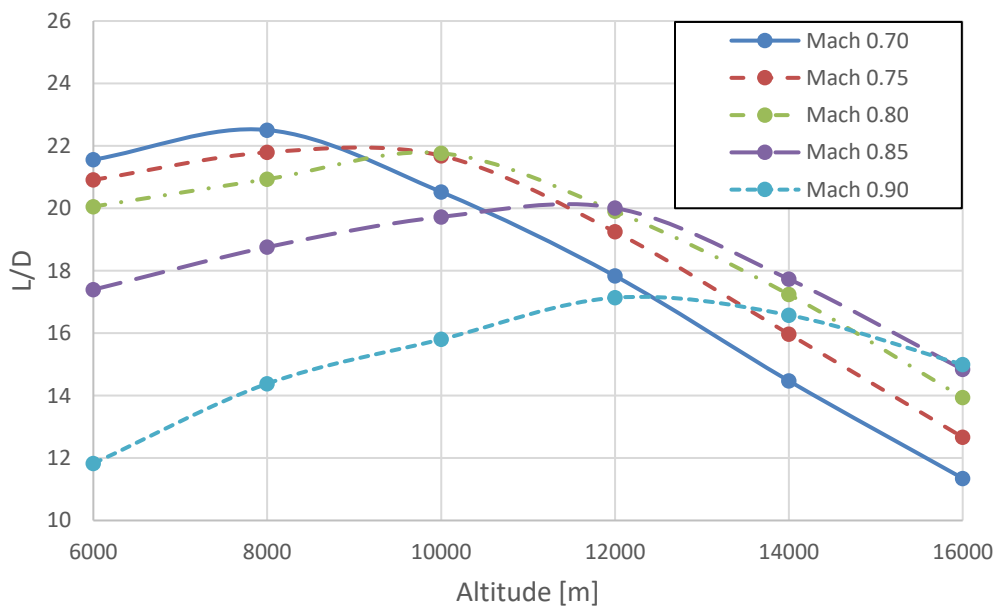


Figure 5-14 Lift to drag ratio at different Mach numbers and altitudes

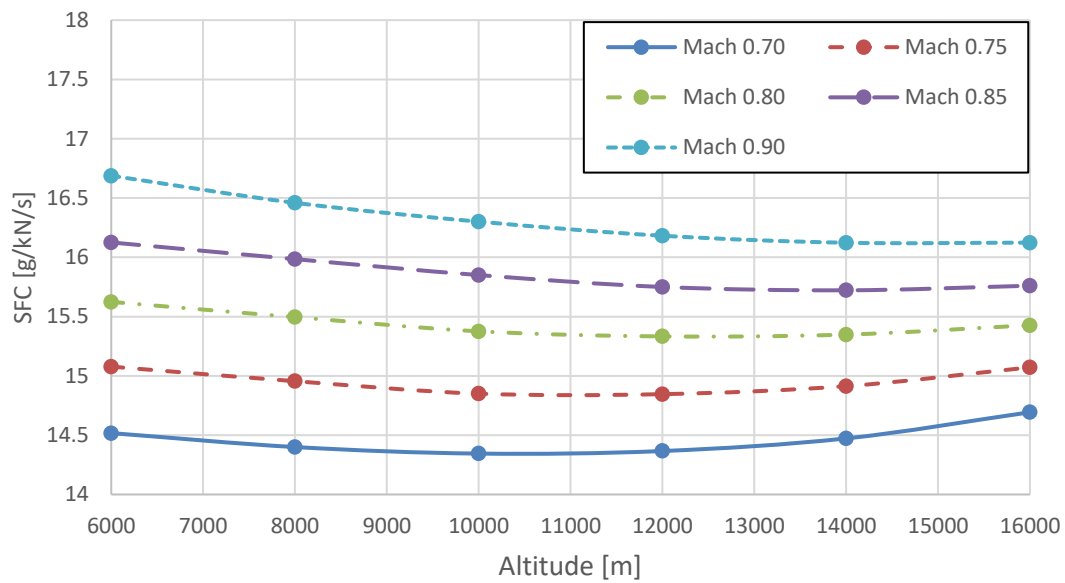


Figure 5-15 SFC at different Mach numbers and altitudes

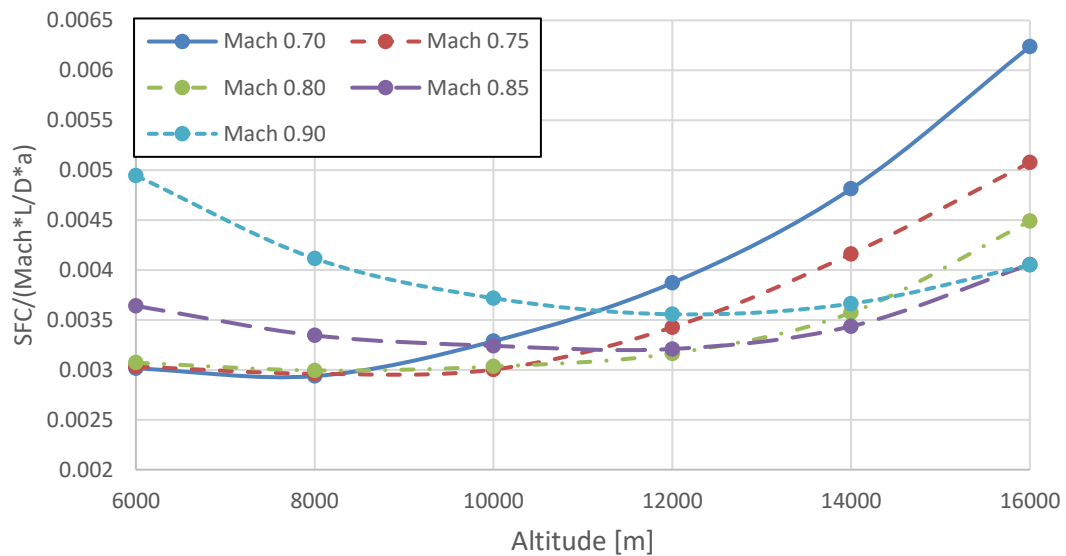


Figure 5-16 Fuel consumption factor

From the above trade-off study, it is possible to determine the most fuel-efficient altitude for a given cruise Mach number. The operation space with the design fuel capability is shown in Figure 5-17. A more comprehensive graphic can be made

for a fixed design to show the most fuel-efficient altitude for different cruise Mach number.

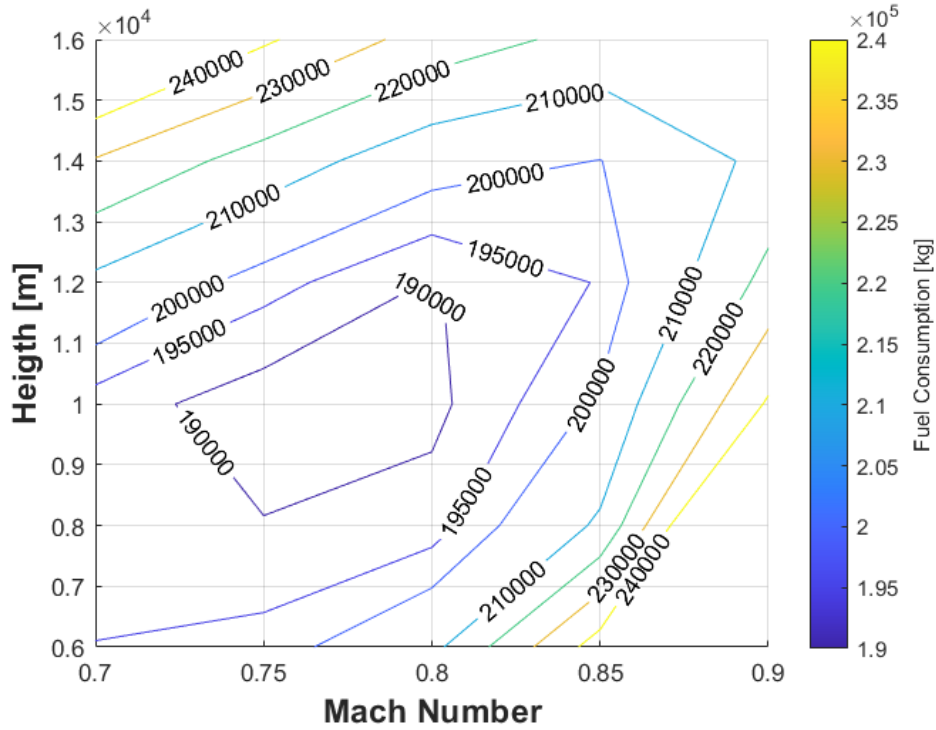


Figure 5-17 Fuel consumption operation space

5.4 Summary

In this chapter, a series of parameter sensitivity analysis have been carried out for a tailless BWB performing a 555-passenger, very long-range mission. these studies is aimed to understand the impact of different design drivers on the characteristics of the BWB aircraft.

At the first part, central body sweep and outer wing sweep have been chosen as the parameters to investigate their influence on the BWB performance at different Mach number. The results show that:

- A higher sweep angle increases the weight of the structure which could rise fuel consumption, but a larger sweep angle increases the lift-drag ratio which is beneficial to reduce fuel consumption, so the final fuel

consumption is a trade-off process. The central body sweep angle plays a decisive role.

- Central body sweep has the greatest influence on the wave drag, and the larger the sweep angle, the smaller the wave drag.
- there is a fuel-efficient central body sweep angle for a specific cruise Mach number. With the increase of Mach number, the optimal central body sweep with the minimum fuel consumption also increase.
- The increase in the sweep angle will cause the Neutral point to move backward, which in turn makes the static margin larger.

The second part is about operational trade-off study for different cruise Mach number and altitude. The result indicates that there is a most fuel-efficient altitude for a specific cruise Mach number. And $\frac{SFC}{L/D * Mach * a}$ can be regarded as the fuel consumption factor to predict the variation trend of fuel consumption with Mach number and cruise altitude.

The above sensitivity analysis research proves the characteristics of strong interdisciplinary coupling of BWB aircraft, and further emphasizes the importance of multi-variable optimisation in the synthesis of BWB aircraft design. The GENUS environment used in this research contains multi-variable optimisation techniques to enable educated users to develop commercially viable BWB aircraft that can be used to conduct optimisation study for the BWB aircraft.

6 Exploration of the BWB Design

6.1 Multivariate Optimisation

There are three optimisation algorithms integrated in the GENUS: a gradient-based algorithm, a genetic algorithm, and a hybrid algorithm, which runs the genetic algorithm optimiser first and then the gradient-based optimiser. The hybrid optimiser has been applied in this research to converge on a high fuel-efficient solution through multivariate optimisation with chosen several geometry and mission variables. The BWB baseline geometry and mission requirement used in this chapter is the same as that used in Chapter 5.

The optimisation objective is formally defined as follows:

$$\text{minimise: } f(\tilde{x}) = \text{fuel consumption} \quad (6-1)$$

Where the input variable vector is defined as:

$$\tilde{x} = (\Lambda_{LE}, b, c, \text{Fuel}_{frac}, \text{Altitude}) \quad (6-2)$$

Subject to:

$$h_0(\tilde{x}) = \frac{|MTOM_{cal} - MTOM_{est}|}{MTOM_{est}} = 0 \quad (6-3)$$

$$h_1(\tilde{x}) = \frac{|Fuel_{cal} - Fuel_{est}|}{Fuel_{est}} = 0 \quad (6-4)$$

$$g_0(\tilde{x}) = l_{LG_{cal}} - l_{LG_{req}} \leq 0 \quad (6-5)$$

$$g_1(\tilde{x}) = h_{struc_{cal}} - h_{struc_{req}} \leq 0 \quad (6-6)$$

$$g_2(\tilde{x}) = X_{rearbar_{cal}} - X_{rearbar_{req}} \leq 0 \quad (6-7)$$

$$g_3(\tilde{x}) = V_{feul_{cal}} - V_{feul_{req}} \leq 0 \quad (6-8)$$

$$g_4(\tilde{x}) = b_{total_{cal}} - b_{total_{req}} \leq 0 \quad (6-9)$$

$$g_5(\tilde{x}) = S_{TO_{cal}} - S_{TO_{req}} \leq 0 \quad (6-10)$$

$$g_6(\tilde{x}) = S_{ld_{cal}} - S_{ld_{req}} \leq 0 \quad (6-11)$$

The convergence criterion is defined as follows:

$$\frac{|f(x_k) - f(x)|}{|f(x_k)|} < \epsilon \quad (6-12)$$

Where:

ϵ is the minimum criteria pre-defined by the user

Additionally, the internal components' longitudinal and lateral positions have checked for clashes, and acceptable CG margins for achieving a longitudinal static margin at all flight conditions within the range $K_n = [-15\%, 15\%]$.

6.2 Impact of Mission Requirement for the BWB Optimisation

The optimisation to get a highest fuel-efficient solution for different Mach number have been done in this section. The variables selected for the optimisation includes geometry parameters: airfoil chords, wing sweeps and wing spans as well as operational parameter: cruise altitude.

The input variable values after optimisation are given in Table 6-1 Input variables after optimisation. The mission performance analysis result of these optimised geometries in different Mach number are compared in the

Table 6-2.

Table 6-1 Input variables after optimisation

Mach	0.70	0.74	0.78	0.82	0.86	0.90
Chord0 (m)	37.62	38.00	41.48	44.21	48.00	52.00
Chord1 (m)	18.49	18.11	21.07	19.62	18.87	18.87
Span0 (m)	12.37	12.57	10.61	10.56	9.93	9.73
Span1 (m)	2.62	2.97	4.28	2.99	3.95	3.44
Span2 (m)	6.00	5.15	6.39	7.48	6.67	6.53
Span3 (m)	19.00	19.32	17.02	18.00	17.89	16.89
Sweep0 (deg)	51.20	54.38	61.92	65.91	69.88	73.38
Sweep1 (deg)	31.32	37.04	42.06	38.02	42.52	41.36
Sweep2 (deg)	35.85	47.10	38.81	43.52	42.26	41.58
Sweep3 (deg)	30.89	37.92	40.20	41.76	43.67	41.69
Altitude (m)	10,640.98	11,494.54	12,003.94	12,673.87	13,160.14	13,742.35

Table 6-2 Mission performance of these optimised geometries at different Mach number

Mach	0.70	0.74	0.78	0.82	0.86	0.90
MTOM (kg)	383,611	387,037	402,802	404,011	422,157	446,140
OEM (kg)	186,252	190,834	196,836	200,082	208,872	212,993
Fuel Consumption (kg)	134,644	133,488	137,251	141,213	149,907	170,431
SFC (g/kN/s)	14.18	14.57	14.98	15.36	15.66	15.96
L/D at Cruise	23.60	23.35	22.45	22.10	21.38	18.69

A top view schematic of the final geometries and the arrangement of the internal components are shown in Figure 6-1. The result shows that the higher wing sweep angle was selected in the final solution as the cruise Mach number increase and aircraft needs higher sweep to reduce the wave drag, despite the structural mass increment as a function of sweep which is shown in Figure 6-2. This is consistent with the parameter sensitivity analysis of sweep angle in Chapter 6: fuel consumption is a trade-off result and increasing sweep angle not

only reduces wave drag but also increases structural weight, which has two opposite effects on fuel consumption. At the low Mach number, wave drag is small and the low sweep angle could bring more benefit from the structure mass reduction. With the increase of Mach number, the influence of wave drag also increases, and even exceeds the influence of structure weight. Therefore, in the case of large Mach number, a larger sweep angle will also be selected in the final solution.

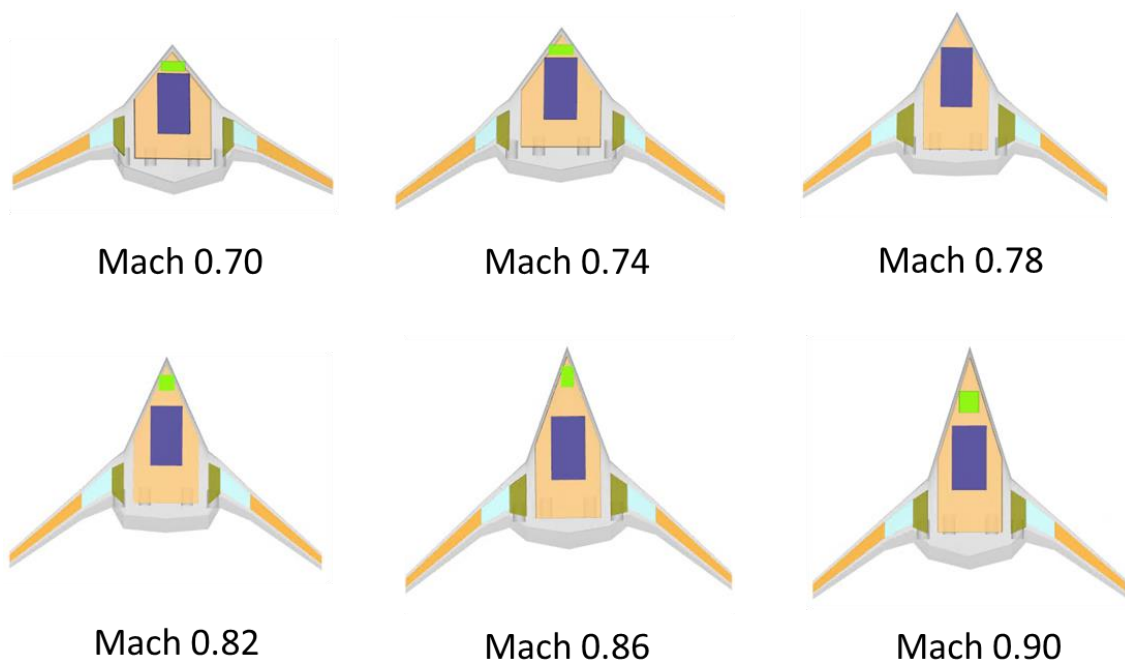


Figure 6-1 A top view schematic of the final geometries at different Mach number

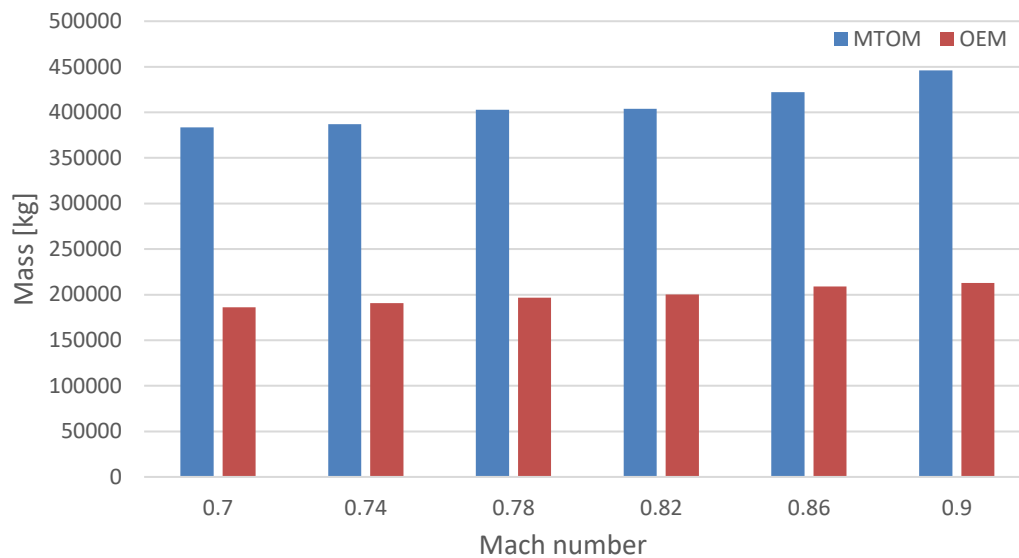


Figure 6-2 MTOM and OEM for the optimised geometries at different Mach number

The overall fuel consumption and mission time at different Mach numbers are plotted in Figure 6-3. It can be noticed that as the cruise Mach number increases from 0.7 to 0.9, fuel consumption first decreases and then increases, and the minimum value is obtained when the Mach number is equal to 0.74. However, the mission time will be significantly increased. It is necessary to define the economic worth of speed before selecting a design cruise Mach number. For example, aeroplane utilisation is directly proportional to speed, and in some cases, a little increase in speed can eliminate the need for a second crew. Thus, the cruise Mach number selection becomes a matter of balancing the increase in MTOM and fuel burn with the reduction in mission time.

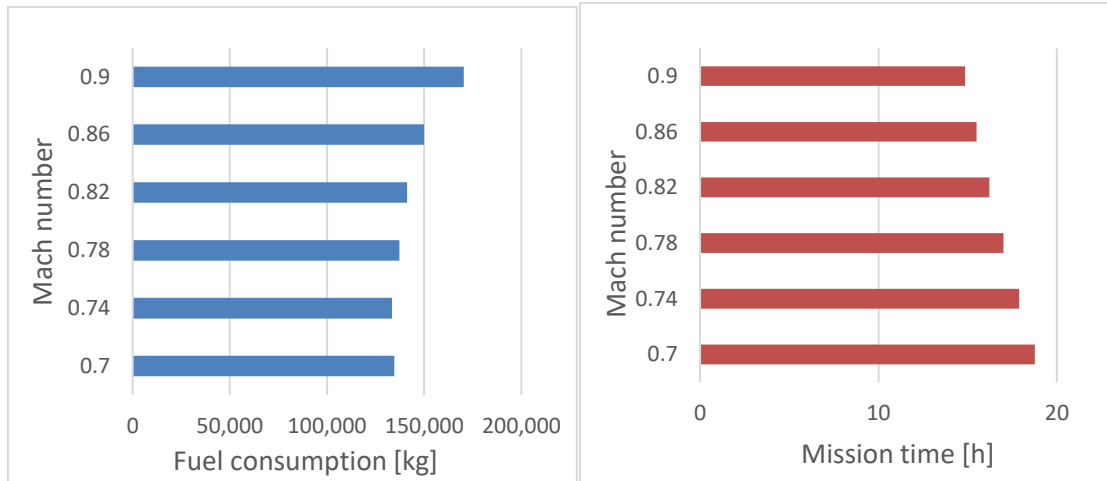


Figure 6-3 Fuel consumption and mission time for different Mach number

In terms of the optimal cruise altitude at different Mach numbers, Figure 6-4 shows higher cruise altitude compared to the initial baseline cruise altitude of 10059m. That is because the lower air density at higher flight altitude improves flight efficiency and reduced the required thrust, which could lead to a lower fuel consumption. In addition, the optimal cruise altitude increases approximately linearly with respect to the Mach number, which is consistent with the conclusion obtained in Chapter 5.3.2. From the previous analysis, it could be known that as the cruise altitude increases, the lift-to-drag ratio first increases and then decreases, and as the Mach number increases, the cruise altitude that obtains the highest lift-to-drag ratio will also increase. SFC also shows the same trend. The cruise altitude to achieve the lowest SFC also increases with the increase of Mach number. The change trend of SFC and lift-to-drag ratio makes the optimal cruise altitude present the trend shown in Figure 6-4.

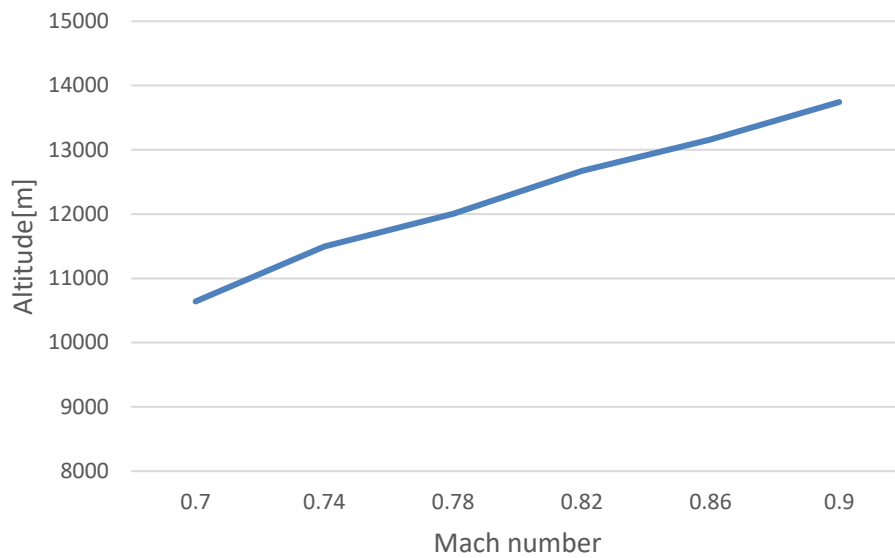


Figure 6-4 Optimal cruise altitude vs cruise Mach number

The L/D and SFC at cruise vs cruise Mach number is plotted in Figure 6-5. SFC increases as the Mach number increases, and the lift-to-drag ratio decreases as the Mach number decreases. It is worth noting that there is a sharp decrease for the L/D when the Mach number increased from 0.86 to 0.90. That is because the optimised geometry at Mach 0.9 has a larger wave drag, which could be clearly seen from Figure 6-6. This problem is caused by the packaging issue. The increase of the sweep angle will increase the overall length of the aircraft, and in order to ensure that the tipback angle during take-off is not less than 15 degrees, the length of the aircraft landing gear also needs to be longer, but too long landing gear will cause the aircraft centre body to be insufficient space to accommodate it, which leads to a packaging problem. For the geometry with a Mach number of 0.9, it should have increased the sweep angle to reduce the wave resistance, but due to the packaging problem caused by the excessively long landing gear, it had to reduce the sweep angle at the expense of increasing the wave drag to obtain a convergence solution. Therefore, we could know that in the case of high Mach number, packaging problem will also become an important factor affecting the shape design of the BWB aircraft.

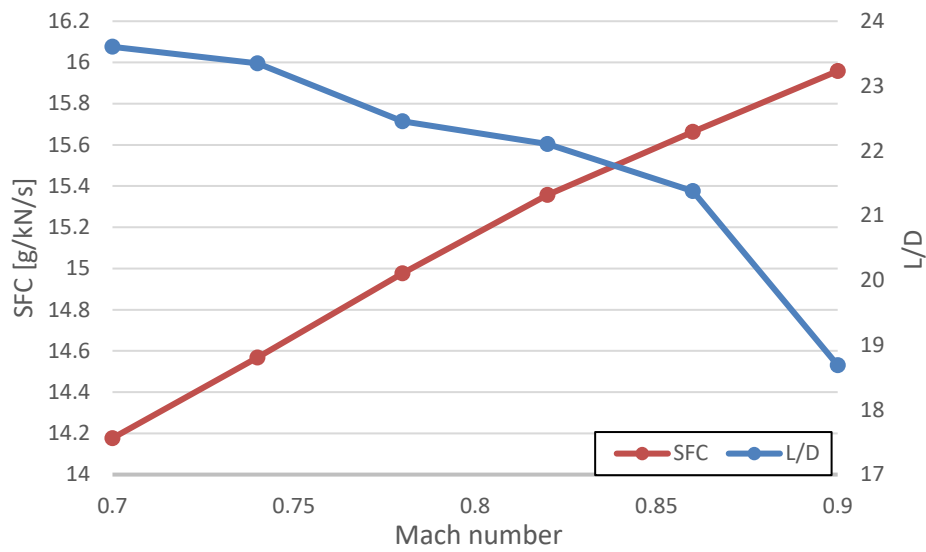


Figure 6-5 SFC and L/D vs cruise Mach number

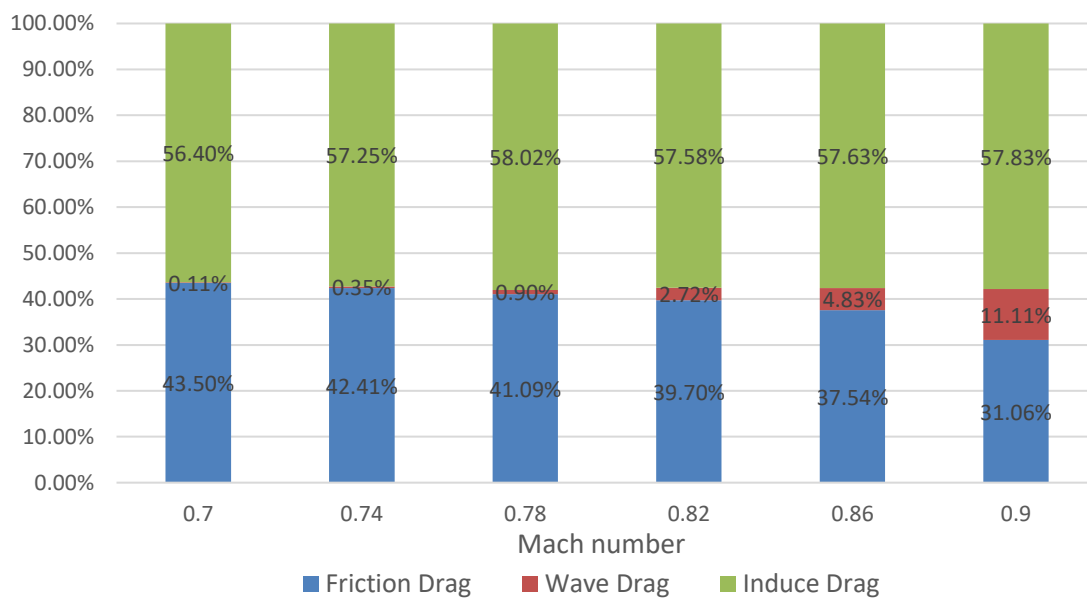


Figure 6-6 Drag build-up for different Mach number

Commercial airlines exist to safely transport passengers to their destinations at the lowest possible cost and with the greatest profit margin. This is the concept upon which the term "productivity" in commercial aviation is based. According to Liebeck, it may be more efficient to design for productivity than aerodynamic efficiency [6]. Eq. (6-13) and Eq. (6-14) quantify productivity and aerodynamic efficiency in terms of the Mach number.

$$Productivity = \frac{Ma * P}{D} \quad (6-13)$$

Where:

Ma is the Mach number.

D is the aerodynamic drag in N .

P is the payload in N .

$$Aerodynamic\ Efficiency = \frac{Ma * L}{D} \quad (6-14)$$

Where:

L is the lift in N .

Figure 6-7 illustrates the relationship between the productivity and aerodynamic efficiency and Mach number. As can be observed from Figure 6-7, productivity and aerodynamic efficiency exhibits a non-linear relationship with Mach number as the Mach number increases, first increasing to a maximum value and then decreasing. Mach 0.74 yielded the highest productivity of 2.8, whereas Mach 0.86 yielded the highest aerodynamic efficiency of 18.6.

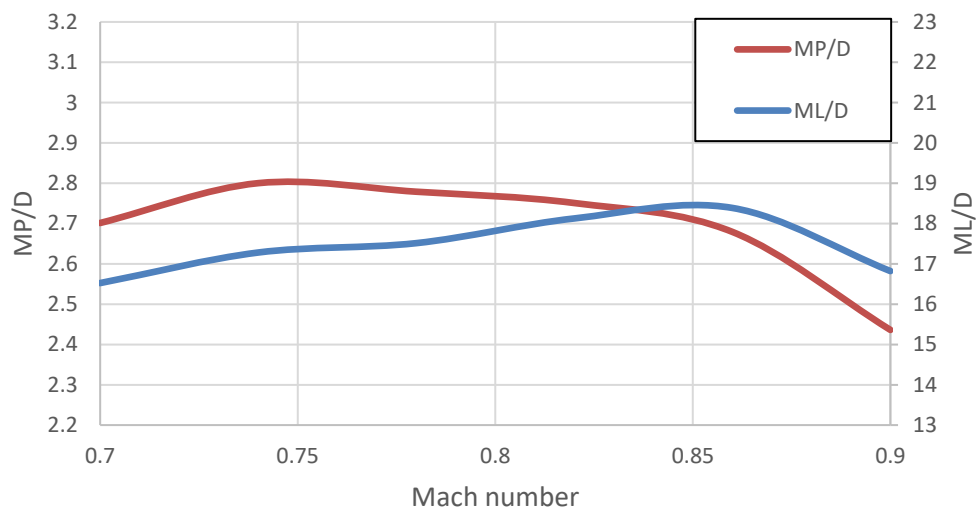


Figure 6-7 Aerodynamic efficiency and productivity at different Mach number

Maximum productivity entails low fuel consumption as a result of minimal drag, resulting in an increase in range or cargo capacity. This demonstrates that maximising aerodynamic efficiency is not always economically advantageous. Therefore, while the natural area rule of BWB configurations makes them ideal for flying at high Mach numbers, high speeds may not always be the best choice. This is because very high speeds may reduce productivity. Thus, design decisions for flight speed must take into account not only aerodynamic efficiency but also productivity, which represents the economic efficiency of the aircraft. In addition, Figure 6-7 indicates that the best aerodynamic performance occurs between Mach 0.84 and Mach 0.88. Above Mach 0.88, the decrease in aerodynamic efficiency may be the result of transonic wave drag.

6.3 Impact of BLI for the BWB Optimisation

The purpose of this section is to explore the impact of the application of boundary layer ingestion technology on the optimisation results of BWB aircraft. From another perspective, it can also be regarded as exploring how the shape of BWB aircraft can be changed to maximize the benefits of BLI technology. In order to achieve this goal, four cases are compared in this section: Case 1 is the baseline BWB geometry described in Chapter 6.2; Case 2 is the optimised geometry of BWB aircraft without BLI; Case 3 is based on the Case 2 with BLI technology; Case 4 is the optimised geometry of BWB aircraft with BLI.

It can be known from the literature review that the benefits of boundary layer ingestion technology mainly depend on the amount of boundary layer ingested by the engine. The boundary layer flowing through the BWB fuselage mainly depends on the flight conditions and the shape of the centre body. Since the mission requirements of these four calculation examples remain the same and the flight conditions remain unchanged, the central body shape of the BWB aircraft is mainly optimised. Therefore, central body sweep, chord 0, chord 1 and span 0 have been selected as the optimisation variables, which determine the size of the central body.

A comparison of the initial geometric variables to the optimised values is given in Table 6-3. The overall performance analysis results are listed in Table 6-4. And

the top views of the case 1, case 2 and case 4 geometries are shown in Figure 6-8. From these results, it can be clearly seen that the length of the central body in Case 4 is greater than that in Case 2. This is because a longer central body length will make the air flow longer through the fuselage, which in turn creates a thicker boundary layer. The thicker boundary layer entering the engine will reduce the average speed of the engine inlet flow and reduce the ram drag of the engine, thereby reducing the fuel consumption of the engine.

Table 6-3 Geometric variables for the four cases

Parameters	Case1: Baseline	Case 2: Optimisation without BLI	Case 3: Optimisation without BLI +BLI	Case 4: Optimisation with BLI
Central body sweep (deg)	63	64.87	64.87	62.19
Chord0 (m)	45	45.21	45.21	46.02
Chord1 (m)	22	19.34	19.34	21.67
Span0 (m)	11	10.24	10.24	10.82

Table 6-4 Mission performance for the four cases

Parameters	Case1: Baseline	Case 2: Optimisation without BLI	Case 3: Optimisation without BLI +BLI	Case 4: Optimisation with BLI
MTOM (kg)	432712	421078	413347	406570
OEM (kg)	208954	203784	201461	205288
Fuel consumption (kg)	161191	155438	149801	138566
SFC	15.32	15.28	15.23	14.97
L/D	21.06	21.86	22.31	23.03

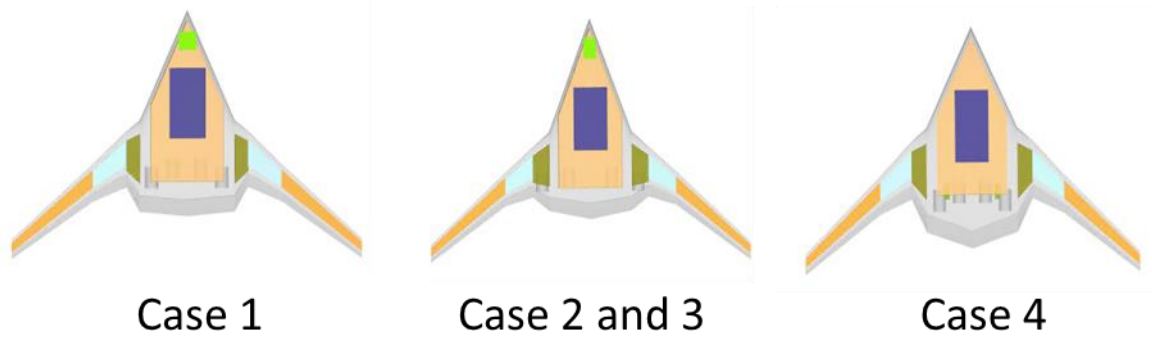


Figure 6-8 A top view for case 1, case 2, case 3 and case 4

However, the longer centre body length means that the aircraft has a higher structural weight, which is not conducive to saving fuel consumption, which can be seen in Figure 6-9. The application of BLI technology leads to a 10.8% reduction in the fuel consumption from 155,439 kg to 138,567 kg. Meanwhile, the increase in the root chord makes Case 4 have a higher structure weight of 205,289 kg, which is 0.8% heavier than Case 2.

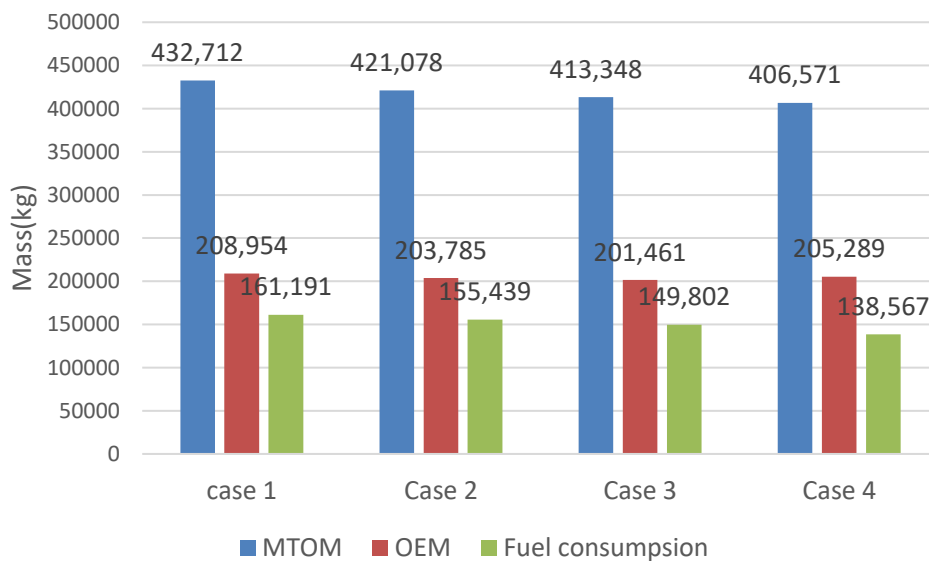


Figure 6-9 MTOM, OEM and fuel mass for the four cases

In addition, a longer airframe will make the landing gear longer, which in turn leads to packaging problems. Therefore, the sweep angle of Case 4 is slightly reduced compared with Case 2. which is likely to have an adverse effect on the aerodynamic efficiency of the design. Figure 6-10 presents the drag buildup for

these four cases. The wave drag in the Case 4 is 0.00105, which is larger than 0.000516 in Case 2. However, Case 4 has a reduced friction drag of 0.0059 compared to 0.0072 for Case 2 due to the application of the BLI.



Figure 6-10 Drag build-up for the four cases

Similarly, the increase in the root chord thickens the boundary layer to reduce the ram drag of the engine, but it also increases the surface area of the fuselage and increases the drag of the aircraft, which has two opposite effects on the fuel consumption. This is also further illustrating the fuel consumption for the BWB aircraft is a trade-off process.

7 Discussion and Conclusion

7.1 Discussion on the Design Methodology and Results

Due to the close-coupled nature of the BWB aircraft, this research developed a multi-disciplinary conceptual design method for the BWB aircraft.

In the developed method, the shape of the BWB aircraft is composed of multiple variables including cross section shapes, platform dimensions, twists, dihedrals, and aerofoils, which provides a high degree of design freedom, and these variables can be used as optimisation variables to optimise the design of the aircraft shape. However, the degree of freedom of the airfoil is not enough. Only the pre-set airfoil can be selected, which means the shape of the airfoil cannot be freely optimised. This can be used as a future development goal to further explore the impact of the airfoil to the BWB aircraft design. In the weight estimation, the semi-empirical method developed for the conventional tube-and-wing aircraft is not suitable for the BWB aircraft due to the difference with the conventional aircraft structure, so the weight estimation method for BWB aircraft cabin developed by Bradley is selected in the GENUS. In this research, several engine analysis tools has been reviewed and the NASA EngineSim was chosen due its flexibility, high accuracy and open source. In the selection of aerodynamic analysis methods, GENUS has implanted a variety of aerodynamic analysis methods. PANAIR is selected for the aerodynamic analysis of the BWB aircraft, because PANAIR can be applied to a wide range of conditions, so as to compare the BWB aircraft with other aircrafts and reduce errors caused by different aerodynamic analysis methods. But PANAIR also has its limitations, PANAIR cannot account for flow separation and vortex breakdown effects at moderate angles of attack, so PANAIR is only suitable for angles of attack between -15 degrees to +15 degrees.

The flight performance analysis is simulated based on Newton's equations of motion, and the entire flight mission is divided into different mission segments, each segment can be set with the initial and end speed and altitude, flight angle, flight time. This approach provides a higher degree of freedom to simulate the

entire flight mission. However, this method requires the designer to plan in advance the speed and altitude requirements that the aircraft needs to reach in each flight segment. For example, in the climb stage, if the set climb angle is too large or the climb time is too short, the aircraft need to reach a too high attack of angle, so that the required lift could be achieved. In addition, the performance analysis needs to use the aerodynamic parameters and engine performance parameters in the specific state of the aircraft. These parameters are linearly interpolated from the matrix obtained in the aerodynamic analysis module and the propulsion system module, which may lead to some errors. In the future development of the GENUS framework, more accurate interpolation methods can be added to reduce such errors.

In this study, the developed method was used to model NASA SUGAR as a validation of this method. The results obtained were compared with the results published by NASA, and the data errors were basically within 10%. In general, the conceptual design methodology for the BWB, as implemented in GENUS, is capable of accurately simulating a BWB aircraft in a variety of aspects, including geometry, aerodynamics, mass estimation, and engine modelling, as well as providing remarkable detail for the various mission segments.

In the selection of optimisation methods, GENUS provides three optimisation methods, gradient-based algorithm, genetic algorithm and hybrid algorithm. The gradient-based algorithm is easy to fall into a local extrema when searching the search space of multimodal distribution, and the genetic algorithm is inefficient and prone to premature convergence. Therefore, the hybrid algorithm is designed, which first uses the genetic algorithm to search for the interval where the extreme value appears, and then uses the gradient-based algorithm to obtain the optimal value. In addition, when using GENUS to optimise the design of a BWB aircraft, the key point is to use enough design parameters to drive all the coupled features, but not too many that the optimisers are overwhelmed; first, runtime can drastically increase. Second, if too many variables are controlling too few outputs, and the correlation between inputs and outputs are weak, the optimiser will just waste effort by varying insignificant parameters. Thus, the

designer not only has to know how many parameters to pick for optimisation, but which ones.

In the BLI analysis module, the flat plate method is used for the simulation of the boundary layer. The height of the boundary layer is only related to the Reynolds number and the flow length. In the actual situation, the thickness of the boundary layer formed by flowing through the airframe is also related to the angle of attack of the aircraft. There is a slight error between the thickness of the boundary layer obtained in this study and the thickness of the boundary layer in the actual situation, which will have a slight impact on the benefits brought by the BLI technology. In view of the fact that this study is aimed at the conceptual design stage of the aircraft, and according to the results of the validation part, the boundary layer thickness obtained by this method has little error compared with the CFD results, so the conclusions and trends obtained by this method are still valid.

7.2 Conclusion

For years, the commercial aviation industry's ongoing attempts to increase aeroplane performance have been combined with a growing desire to reduce its environmental impact. These goals necessitate game-changing solutions, such as new aircraft configurations, technology, and procedures. The Blended Wing Body (BWB) design and boundary layer ingest (BLI) technology are two of the most promising among the several novel alternatives. The BWB configuration philosophy entails reducing the total external geometry of the aircraft to the only subsystems which contribute to lift. The highly coupled configuration, on the other hand, poses a packaging issue that, if not addressed early in the design phase, could become a bigger burden later. BLI technology is also an emerging technology. Many studies have shown that the introduction of BLI technology can bring significant gains to aircraft. But the degree of gain is still uncertain, because most previous studies have separated the propulsion system from the aircraft as an independent individual. This research is to treat the BWB aircraft and BLI technology as a whole to develop a multidisciplinary design, analysis and

optimisation tool, which can carry out in-depth studies founded on physics to limit the design space and thus reduce future development costs.

A multidisciplinary design analysis and optimisation BWB model has been developed based on the GENUS aircraft conceptual design environment. The general GENUS framework has been verified to be functional correctly; a correct interaction between modules and the different optimisation algorithms has also been evaluated. A semi-empirical weight estimation method appropriate to BWB has been applied in the GENUS and a robust and flexible packaging module has been developed to locate the components and resolve the clashes between the components.

A method for sizing and simulating the performance of a BLI propulsion system has been integrated into GENUS. Through a mass flow matching process, boundary layer simulations have been coupled with traditional 1-D gas turbine performance methodologies to forecast the performance of a propulsion system under a variety of flight conditions.

The comparative case studies of BWB aircraft and conventional tube-and-wing aircraft in three different classes have been conducted in this research. The result presents that the BWB is more aerodynamically and structurally efficient than the tube-and-wing aircrafts, especially at larger aircraft sizes. The mass estimations predict the operating empty weight to be higher for a BWB than for a tube-and-wing aircraft at small size (200 pax class) and become lower when the number of passengers exceeds 300. Together with the higher aerodynamic efficiency, this results in lower fuel burn with values as high as 10.4% for a very long-range 555-passenger aircraft. The application of BLI technology further expands the advantages of BWB, and the benefit from BLI increase with aircraft size, such as from 2.9% to 3.9% increase in lift to drag ratio and from 3.9% to 6.5% reduction in fuel consumption compared to the initial BWB aircraft.

A series of parameter sensitivity analysis have been carried out through the developed BWB design model to understand the impact of different design drivers on the characteristics of the BWB aircraft. From the result, it can be seen that performance of BWB aircraft shows a high dependence on the central body

sweep angle. A higher sweep angle not only increases the lift to drag ratio but also increases the weight of the structure, which has two opposite effects on reducing fuel consumption. And as the Mach number increases, the most fuel-efficient sweep angle also increases. Thus, the geometry design of the BWB is a trade-off process according to the cruise Mach number. In terms of the operational trade-off study, there is a fuel-efficient altitude for a specific cruise Mach number.

The highest fuel-efficient solution for different Mach number has been obtained through the optimisation based a BWB baseline geometry. It can be found that the geometry at Mach 0.74 has the lowest fuel consumption, but the one at Mach 0.86 has the highest productivity. Hence, the design decision for the BWB is a trade-off process. With the application of BLI, increasing the length of the central body is conducive to maximizing the benefits of BLI technology. The optimised geometry with boundary layer ingestion leads to a 10.8% reduction in the fuel consumption but an 0.8% higher structure weight compared to the optimised geometry without boundary layer ingestion.

7.3 Contributions of Knowledge

This research provides the following significant contributions to knowledge:

1. A flexible conceptual design methodology that supports the iterative and unique process of Blended Wing Body aircraft design with Boundary Layer Ingestion technology. This method couples the multi-fidelity aerodynamic analysis model, the BWB specific structural quality module and the BLI analysis model into GENUS conceptual design environment for rapidly designing and exploring the BWB aircraft's design space.
2. Through comparative case studies of BWB aircrafts and conventional tube and wing aircrafts in different sizes, the applicable scope of BWB airplanes and the degree of benefit of boundary layer ingestion technology to airplanes of different sizes are obtained.
3. Through the exploration of the design space of the BWB aircraft by sensitivity analysis and optimisation results analysis, the impact of geometric variables,

mission requirement and application of BLI to the BWB aircraft performance are obtained, which provides ideas and references for the subsequent design of the BWB aircraft.

7.4 Future Work

While the author of thesis believes that the methodologies used in this thesis are sufficient to show the usefulness of the method and to substantiate the contributions of the thesis, it is clear that the methodology for the BWB and BLI modelling is not yet where it needs to be. This part will propose several future work recommendations for the BLI modelling and BWB conceptual design methodology respectively.

BLI modelling

- Higher fidelity component modelling should be introduced into the BLI model to estimate the effect of the BLI more accurately. For example, CFD method to model the boundary layer performance; better estimates for inlet recovery and performance at mass flow ratio, including lip losses, pre-entry flow separation losses; estimates of external cowl, spillage, and interference drags for different inlets and parameterized for multiple configurations and so on.
- Expand the scope of application of the BLI model. The BLI module can be combined with the distributed propulsion system and the hybrid electric propulsion system to further explore the benefits of the BLI technology. In addition, BLI technology can also be introduced to conventional aircraft, where the engine is placed on the tail, so that the performance of conventional aircraft has also been improved.

BWB conceptual design methodology

- More flexible geometry parameterisation should be implemented in the geometry model to explore a more suitable airfoil for BWB aircraft
- Higher-fidelity methods should be implanted for more accurate analysis such as CFD method for the aerodynamic module and FEA method for the mass breakdown module.

- According to literature review, low emissions (CO₂ and NO_x) and low noise are also the advantages of BWB aircraft over traditional aircraft. Emission module and noise analysis module can be developed as special modules into the BWB conceptual design methodology.
- Gradient algorithm and genetic algorithm are very classic optimisation methods, but they also have an unavoidable shortcoming. Some more efficient optimisation methods can be introduced to obtain optimisation results faster.

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APPENDIX

Appendix A Atmospheric Properties

Atmospheric properties at altitude follow the US 1976 International Standard Atmosphere (ISA). The heat capacity ratio of air, γ , interpolates data from ESDU 00.01.08[139].

$$T = T_n + \lambda_n(h - h_n) \quad (\text{A-1})$$

$$P = P_n \frac{T_n^{\frac{g}{R\lambda_n}}}{T} \quad \lambda_n \neq 0 \quad (\text{A-2})$$

$$P = P_n e^{\frac{-g(h-h_n)}{RT_n}} \quad \lambda_n = 0 \quad (\text{A-3})$$

$$\rho = \frac{P}{RT} \quad (\text{A-4})$$

$$a = \sqrt{\gamma RT} \quad (\text{A-5})$$

$$\mu = \mu_0 \left(\frac{T}{T_n} \right)^{1.5} \frac{T_0 - 110.4}{T + 110.4} \quad (\text{A-6})$$

a = Speed of sound

g = Gravitational acceleration (= 9.80665 m/s)

h = Altitude, m

P = Pressure at altitude, Pa

P_n = Pressure at start of atmosphere level n

R = Specific gas constant of air (= 287.0579 J / kg·K)

T = Temperature at altitude, K

T_n = Temperature at start of atmosphere level n

γ = Specific heat capacity ratio of air

λ_n = Temperature lapse rate, K/m

μ = Dynamic viscosity of air at altitude, $\text{Pa}\cdot\text{s}$

μ_0 = Dynamic viscosity of air at sea level (= 0.0000181206 $\text{Pa}\cdot\text{s}$)

ρ = Density of air at altitude, kg/m^3

Table A-1 Atmosphere layers

n	Altitude, $h_n(\text{km})$	Temperature, $T_n(\text{K})$	Lapse Rate, $\lambda_n(\text{K}/\text{km})$	Pressure, $P_n(\text{Pa})$
0	0	288.15	-6.5	101325
1	11	216.65	0	22632.1
2	20	216.65	1	5474.89
3	32	228.65	2.8	868.02
4	47	270.65	0	110.91
5	51	270.65	2.8	66.94
6	71	214.65	2	3.96

Appendix B Mass Breakdown Method

This appendix contains the set of Class II mass estimation methods used for BWB design. Several methods have been employed during the course of this research. The geometric definition for the BWB configuration in this research is shown in Figure B-1. The fuselage of this BWB configuration consists of the cabin and the aft body.

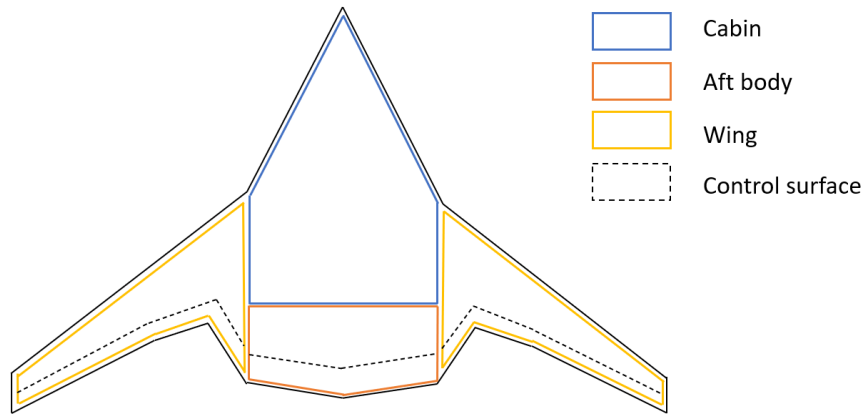


Figure B-1 the geometric definition for the BWB configuration

B.1 Structural Components

Wing

$$M_{wing} = C_1 \left[\frac{bS}{\cos \Lambda_{0.25}} \left(\frac{1 + 2\lambda}{3 + 3\lambda} \right) \left(\frac{MTOM}{S_{wing}} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \quad (\text{B-1})$$

Where:

C_1 is 0.028 for long range transport aircraft.

b is the wing span in m .

$MTOM$ is maximum take off mass in kg .

τ is the average thickness to chord ratio.

λ is the wing taper ratio.

$\Lambda_{0.25}$ is the quarter chord sweep angle in deg.

S_{wing} is the wing area m^2 .

n is design normal acceleration factor.

V_D is the design dive speed in m/s EAS.

Fuselage

$$M_{fuselage} = M_{cabin} + M_{aftbody} \quad (B-2)$$

$$M_{cabin} = 1.8MTOM^{0.167}A_{cabin}^{1.06} \quad (B-3)$$

$$M_{aftbody} = (1.0 + 0.05NFE) * 0.53 * S_{aftb} * MTOM^{0.2} * (0.5 + TR_{aftb}) \quad (B-4)$$

Where:

NFE is number of fuselage mounted engines

S_{aftb} is area of the aft body m^2 .

TR_{aftb} is taper ratio of the aft body, which is calculated with the following equation:

$$TR_{aftb} = \frac{(1.0 - RSPSOB) * \frac{XLW}{RSPSOB}}{(1.0 - RSPCHD) * XL} \quad (B-5)$$

Where:

$RSPCHD$ is percent chord of the HWB fuselage rear spar at the fuselage.

$RSPSOB$ is percent chord of the HWB fuselage rear spar at the side of body. By default $RSPSOB = RSPCHD$.

XL is total fuselage length m .

XLW is fixed length of side wall m .

The area of the aft body is calculated with the following equation.

$$S_{aftb} = S_{fuselage} - A_{cabin} \quad (B-6)$$

Where:

A_{cabin} is passenger cabin floor area m^2 .

$S_{fuselage}$ is fuselage planform area, which is defined by following equation:

$$S_{fuselage} = WF * \frac{XL + XLW}{2RPSOB} \quad (B-7)$$

WF is the maximum fuselage width in m .

Additionally, a technology reduction factor can be applied to structural mass components, especially to the wing and fuselage, to account for composite materials. A conservative estimate for subsonic aircraft is a structural mass reduction between 10% and 15% due to advanced materials and manufacturing practices.

Landing Gear

$$M_{noseLG} = 20 + 0.10 * MTOM^{0.75} + 0.000002 * MTOM^{1.5} \quad (B-8)$$

$$M_{mainLG} = 40 + 0.16 * MTOM^{0.75} + 0.019 * MTOM + 0.000015 * MTOM^{1.5} \quad (B-9)$$

B.2 Propulsion System Items

Installed Powerplant

For a non-afterburning engine:

$$M_{eng} = 1.2 * 14.7T_0^{1.1}e^{-0.045BPR} \quad (B-10)$$

For an afterburning engine:

$$M_{eng} = 1.2 * 11.1T_0^{1.1}Ma_{lim}^{0.25}e^{-0.81BPR} \quad (B-11)$$

Miscellaneous Propulsion Systems

The miscellaneous propulsion systems weight is composed of engine controls and starters.

$$M_{engControl} = 0.26 * NFE * T_0^{0.5} \quad (B-12)$$

$$M_{engStarter} = 11.0 * NFE * Ma_{lim}^{0.32} * D_{eng}^{0.5} \quad (B-13)$$

Where:

T_0 is the maximum thrust of each engine at sea level kN .

Ma_{lim} is the maximum Mach number

D_{eng} is the average diameter of each engine m .

Fuel System

$$M_{fuelSys} = 1.07 * M_{fuel}^{0.58} * NFE^{0.43} * Ma_{lim}^{0.34} \quad (B-14)$$

Where:

M_{fuel} is aircraft maximum fuel capacity kg .

B.3 Systems and Equipment Items

Surface Controls

$$M_{sc} = 1.1 * S_{conSurface}^{0.6} * MTOM^{0.6} * Ma_{lim}^{0.52} \quad (B-15)$$

Where:

$S_{conSurface}$ is total movable wing surface area including flaps, elevators etc m^2 .

Auxiliary Power Unit

$$M_{APU} = 54 * S_{fuselage}^{0.3} + 5.4 * N_{passenger}^{0.9} \quad (B-16)$$

Instruments

$$M_{Instruments} = 0.48 * S_{fuselage}^{0.57} * M_{lim}^{0.5} * (10 + 2.5 * N_{crew} + 1.25 * NFE) \quad (B-17)$$

Hydraulics

$$M_{hydraulic} = 0.57 * (S_{fuselage} + 0.27 * S_{wing}) * (1 + 0.03 * 0.04 * NFE) \quad (\text{B-18})$$

$$* \left(\frac{3000}{HYDPR} \right)^{0.35} * (1 + 0.04 * VARSWP) * Ma_{lim}^{0.33}$$

Where:

HYDPR is hydraulic system pressure, *Pa*. The default value is 3000.

VARSWP is wing variable sweep weight penalty factor ranging from 0.0 for fixed-geometry wing to 1.0 for full variable-sweep wing.

Electrical

$$M_{electrical} = 92 * XL^{0.4} * WF^{0.14} * N_{fuse}^{0.27} * NFE^{0.69} * (1 + 0.044 \quad (\text{B-19})$$

$$* N_{crew} + 0.0015 * N_{passenger})$$

Avionics

$$M_{avionics} = 15.8 * Range^{0.1} * N_{crew}^{0.7} * S_{fuselage}^{0.43} \quad (\text{B-20})$$

Furnishings and Equipment

$$M_{furnishings} = 127 * N_{crew} + 112 * NPF + 78 * NPB + 44 * NPT + 2.6 \quad (\text{B-21})$$

$$* (1 - CARGF)$$

$$* \left[\frac{A_{cabin} * (WF + DF * NBAY)}{WF} + WF * DF * (1 \right.$$

$$\left. + \frac{1}{\cos \Lambda_{fuse}}) \right]$$

Where:

CARGF is cargo aircraft floor factor, where 0.0 is for passenger transport and 1.0 is for military cargo transport.

Λ_{fuse} is sweep angle of the passenger cabin, *rad*.

Air Conditioning

$$M_{AC} = \left(3.2 * (S_{fuselage} * DF)^{0.6} + 9 * N_{passenger}^{0.83} \right) * Ma_{lim} + 0.075 \quad (\text{B-22})$$

$$* M_{avionics}$$

Anti-Icing

$$M_{AI} = \frac{b}{\cos \Lambda_{0.25}} + 3.8 * D_{eng} * NFE + 1.5 * WF \quad (\text{B-23})$$

B.4 Operating Items

The operating items include the crew and baggage, unusable fuel, engine oil, passenger service, and cargo containers. The weight of the crew and baggage and cargo containers could be input by users.

Engine Oil

$$M_{oil} = 0.082 * NFE * T_0^{0.65} \quad (\text{B-24})$$

Passenger Service

$$M_{SRV} = (5.164 * NPF + 3.846 * NPB + 2.529 * NPT) * \left(\frac{Range}{Ma_{lim}} \right)^{0.225} \quad (\text{B-25})$$

Appendix C Conceptual Design Process for a BWB Configuration in GENUS

This section will introduce the conceptual design efforts towards establishing a 555-passenger BWB configuration.

C.1 Mission Design

The requirements are loosely based on the performance characteristics of an A380 aircraft. The top-level requirements that are used for this study are listed in Table C-1 and the mission segments are shown in Figure C-1.

Table C-1 Top-level requirements

Requirement	Value
Seat Capacity	9+80+466, 3 classes configuration
Cruise Range (km)	>14,000
Cruise Mach Number	0.82
Cruise Altitude (m)	10,059
Maximum take-off distance (m)	2,500
Maximum landing distance (m)	2,000
Passenger and baggage mass (kg)	83+32

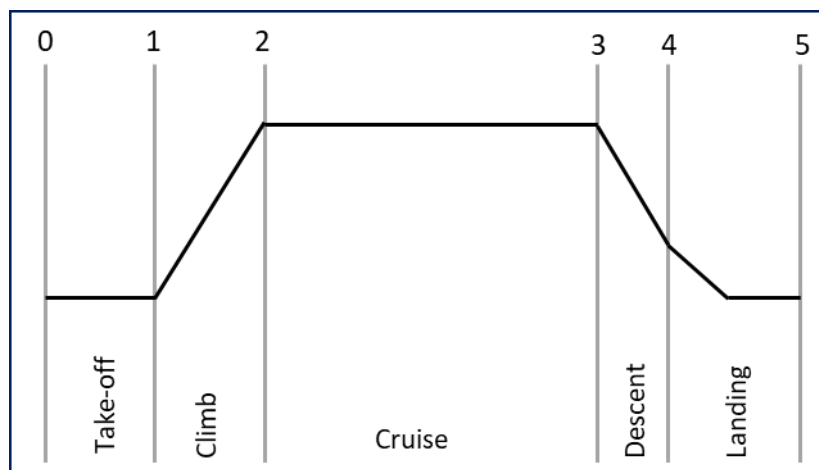


Figure C-1 Typical mission profile for this design

C.2 Initial Design Assumption

The assumptions made for the preliminary sizing are listed in Table C-2. This BWB configuration features redundant elevons located at the trailing edge of the central body and outer wing. These control surfaces are multi-functional, including trim, longitudinal and lateral control and lift increments. In addition, there is no high-lift device arranged for this design. An initial parametric analysis has been carried out with low-fidelity tools and estimates in order to obtain wing and thrust loadings. The initial design point is shown in Figure C-2, with a wing loading $3,470 \text{ N/m}^2$ and a thrust loading 0.26. The take-off mass has been estimated as approximately 500,000 kg, resulting in a wing area of $1,440 \text{ m}^2$ and a sea level thrust of 1,275 kN.

Table C-2 Initial assumptions for the preliminary sizing

Parameter	Value
Maximum lift coefficient clean	0.9
Stall speed clean (m/s)	80
Stall speed take-off (m/s)	70
Stall speed landing (m/s)	70
Aspect ratio	4.0
Maximum lift-to-drag ratio	24
Cruise lift-to-drag ratio	20.8

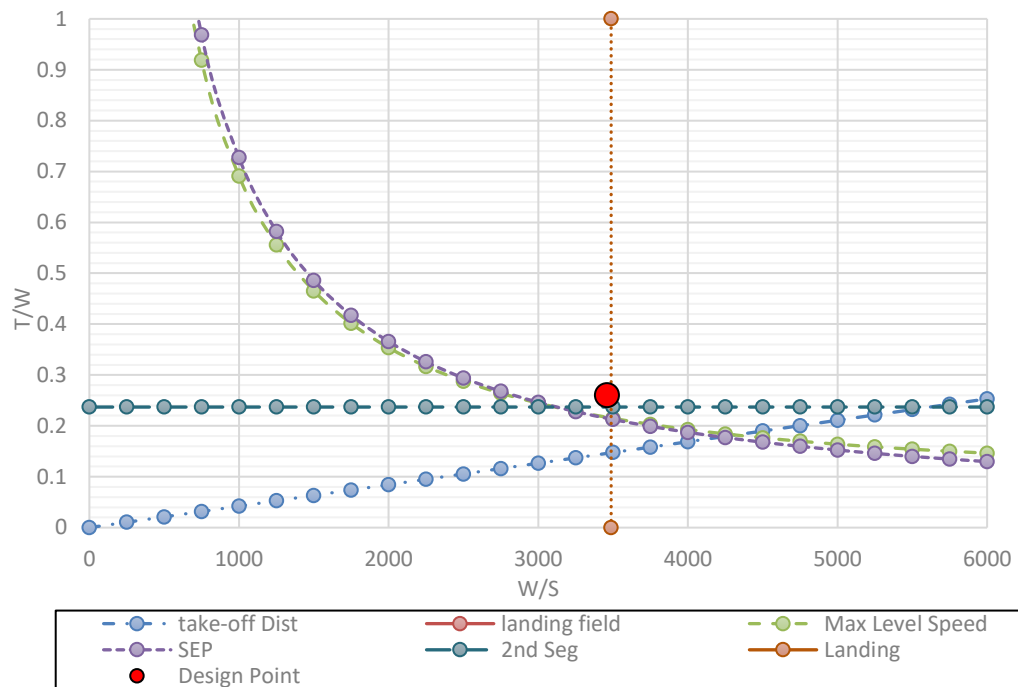


Figure C-2 Initial parameter analysis

A high sweep angle at the central body (fixed at 63°) allows for a longer configuration which will reduce wave drag and allow for a better arrangement of the components. The sweep angle at the outer wing was assumed to be 48° which could reduce the structure mass. the central aerofoil of 48 m chord and 15% thickness is assumed according to the packaging issue. A simple top-view schematic of a packaged configuration is shown in Figure C-3.



Figure C-3 Top view schematic of the initial design

C.3 Design study

This section describes the trade studies for aerofoils and the engine modelling process.

C.3.1 Aerofoil Selection

This section aims to explore the influence of different aerofoils on fuel consumption in order to select the optimal aerofoil for this design. This study were carried out at a fixed sweep angle and thickness-to-ratio. The results for the fuel consumption for different aerofoil are shown in Figure C-4. For a fixed sweep angle, the NACA4 outperforms all other choices and results in a configuration with a lowest fuel consumption. Therefore, NACA4 aerofoils were selected in this design case.

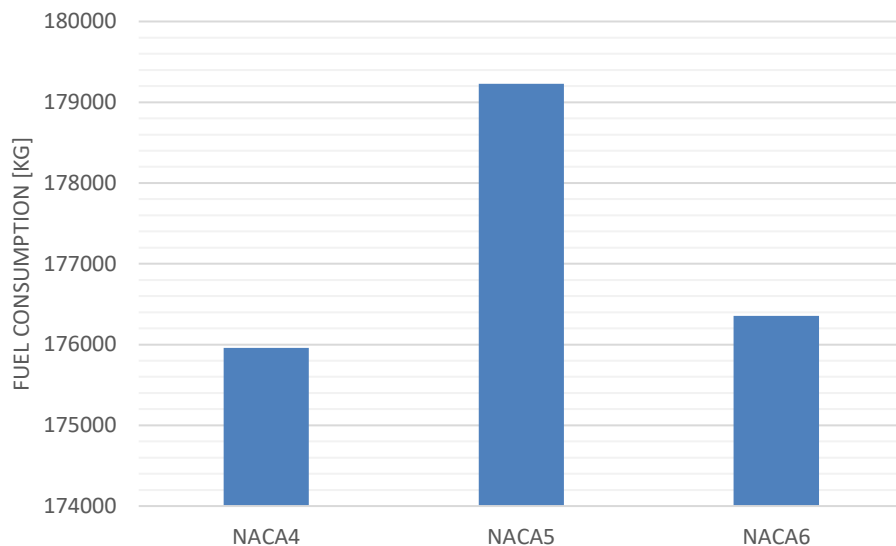


Figure C-4 Mission fuel consumption for different aerofoils

C.3.2 Engine Selection

The mission requirements of this design are the same as A380, so the same engine as A380 is selected. Hence, the Trent-900 engine with sea level maximum thrust of 366 kN and cruise SFC of 15.7 g/(kN·s) has been modelled by the EngineSim, which has been integrated in the GENUS. The required inputs for this engine model are listed in Table C-3.

Table C-3 Required inputs for the Trent-900 engine model

Parameter	Value
Fan bypass ratio	8.5
Fan diameter (m)	2.95
Fan pressure ratio	1.3
Compressor pressure ratio	27
Turbine exit temperature (K)	1580 K
Inlet efficiency	0.98
Fan efficiency	0.89
Compressor efficiency	0.87
Burner efficiency	0.98
Turbine efficiency	0.91
Nozzle efficiency	0.98

The surrogate matrices of the thrust and SFC at different Mach number and altitude generated in GENUS are shown in Figure C-5 and Figure C-6.

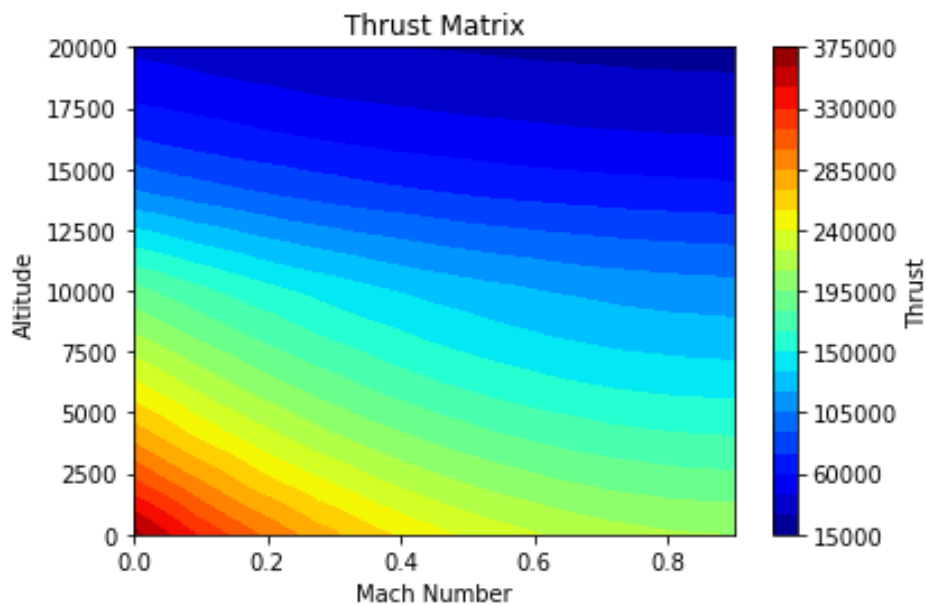


Figure C-5 Surrogate matrix of the thrust at different Mach number and altitude

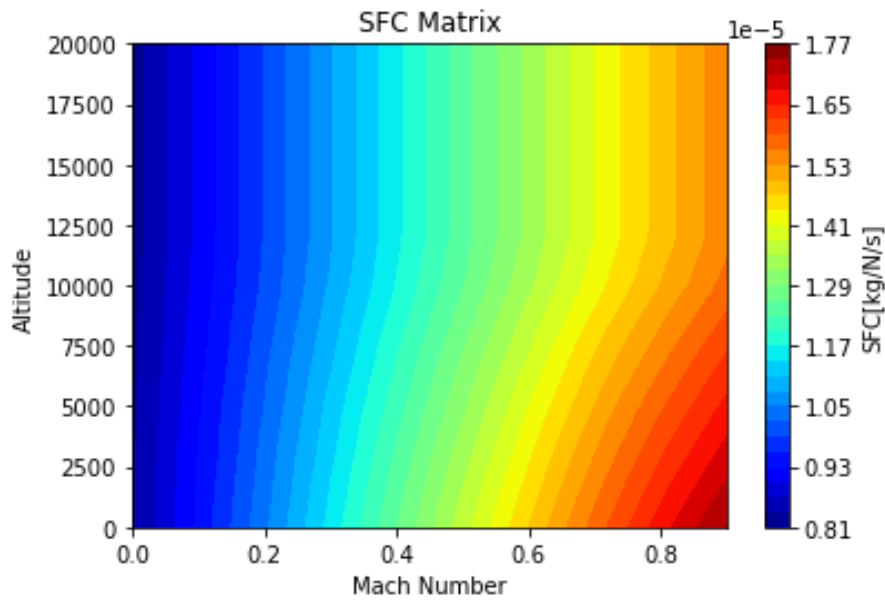


Figure C-6 Surrogate matrix of SFC at different Mach number and altitude

C.4 Multivariate Optimisation

The GENUS genetic and gradient-based optimisers have been used in hybrid mode to converge on a low fuel mass solution through multivariate optimisation. The variables selected for the optimisation includes geometry parameters: airfoil chords, wing sweeps and wing spans. The constraints are selected to ensure the aircraft can fulfil the range requirement, packaging requirement, take-off and landing distance requirement. The optimisation settings and results are listed in Table C-4.

Table C-4 The optimisation settings and results for this design

	Parameters	Initial values	Lower boundary	Upper boundary	Optimisation results
Objective	Fuel consumption (kg)	175,960	-	-	148,268
Variables	Chord0 (m)	48	42	50	44.21
	Chord1 (m)	22	18	26	19.62
	Span0 (m)	13	9	15	10.56
	Span1 (m)	4.5	2	7	2.98
	Span2 (m)	6.0	4	8	7.48
	Span3 (m)	15.25	10	20	18
	Sweep0 (deg)	63	55	70	65.91
	Sweep1 (deg)	38	30	45	38.02
	Sweep2 (deg)	38	30	45	43.52
	Sweep3 (deg)	38	30	45	41.76
Constraints	MTOM error (=0)	0.048	-	-	0.001
	Fuel error (=0)	0.001	-	-	0.001
	Total span (m)	77.42	-	80	78.69
	Take-off distance (m)	2007.23	-	2,500	1880.52
	Landing distance (m)	1686.54	-	2,000	1466.17

Figure C-7 gives the comparisons of low-boom configuration and baseline configuration from front view and top view. The optimised configuration can be found to have a small increase in sweep angle to reduce wave drag, and a smaller central body and longer outer wing.

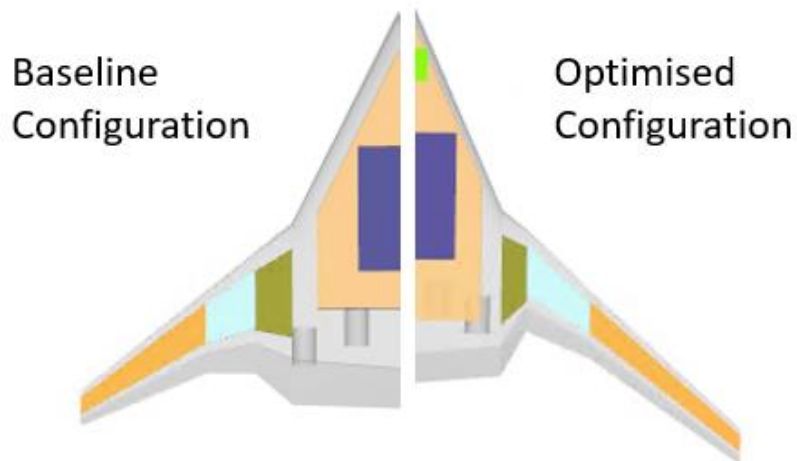


Figure C-7 Comparison of the baseline and optimised configuration

The convergence history of objective function fuel consumption is plotted in Figure C-8. The genetic algorithm converges very quickly. If the objective function is not changed in ten generations, it is regarded as converged.

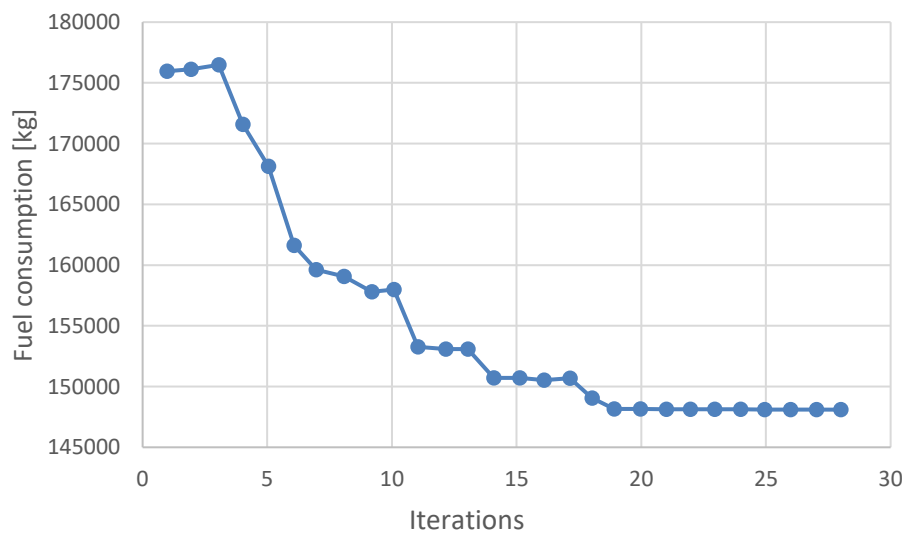


Figure C-8 Convergence history of the objective function

C.5 Final Solution

This part will give a final design solution of a high fuel-efficient, 555-passenger BWB design case to demonstrate the ability of the design model developed in GENUS.

MISSION

Design Mach number:	0.82
Design altitude:	10,059 m
Design payload:	63,825 kg
Design range:	14,000 km
Estimated MTOM:	413,949 kg
Fuel:	148,268 kg
Take-off distance:	2,500 m
Landing distance:	2,000 m

GEOMOTRY

See Figure C-9

Gross area:	1,137.96 sqm
Total span:	78.69 m
Aspect ratio:	5.345
MAC:	23.77 m
Kinks:	3
Chord0:	44.21 m
Chord1:	19.62 m

Chord2:	12.0 m
Chord3:	8.0 m
Chord4:	4.23 m
Span0:	10.56 m
Span1:	2.98 m
Span2:	7.48 m
Span3:	18.0 m
Sweep0:	65.91 deg
Sweep1:	38.02 deg
Sweep2:	43.52 deg
Sweep3:	41.76 deg
Dihedral0:	0.0 deg
Dihedral1:	1.5 deg
Dihedral2:	3.0 deg
Dihedral3:	3.0 deg
Incidence0:	0.0 deg
Incidence1:	0.7 deg
Incidence2:	3.0 deg
Incidence3:	1.02 deg
Incidence4:	2.43 deg
Aerofoil section:	
Aerofoil0:	NACA 4016

Aerofoil1:	NACA 4005
Aerofoil2:	NACA 4009
Aerofoil3:	NACA 4008
Aerofoil4:	NACA 4008

Flaps:

Type:	Plain flap
Control chord ratio0:	0.08
Control chord ratio1:	0.18
Control chord ratio2:	0.25
Control chord ratio3:	0.25
Control chord ratio4:	0.25
Movement:	up 20 deg – down 20 deg

Cabin:

Central length:	34.56 m
Width:	16.25 m
Wall length:	16.39 m
Height:	2.2 m

Nose Undercarriage

Type:	Forward retracting
Length:	5.06 m
X position:	4.42 m
Y position:	0.0 m

Main Undercarriage:

Type:	Forward retracting
Length:	4.92 m
X position:	34.5 m
Y position:	5.85 m

AERODYNAMICS

Lift curve slop	See Figure C-10
Zero lift drag coefficient	See Figure C-11
Zero Moment coefficient	See Figure C-12
Induce drag coefficient factor	See Figure C-13
Aerodynamic centre position	See Figure C-14
Spanwise lift distribution	See Figure C-15

POWER PLANT

Type:	4×Trent 900 high-bypass turbofan engine
Design point:	15000m, Ma 0.82
Design point thrust:	67.2kN
SLS thrust:	365.5kN
Bypass ratio:	8.5
Maximum engine diameter:	3 m
Trust vs Mach & Alt:	See Figure C-16
SFC vs Mach & Alt:	See Figure C-17

PACKAGING & CG

MTOM: 413,949 kg	CG: 29.72 m
MZFM: 264,129 kg	CG: 27.40 m
OEM: 201,414 kg	CG: 28.60 m
Mass Breakdown and C.G.	See Table C-7
Nose LG Min load:	13.07%
Nose LG Max load:	19.24%
Packaging Layout	See Figure C-18

Performance

The performance gives the flight conditions at different segments. Each segment is analysed by dividing it into dozens of small segments.

Table C-5 Performance results

Flight condition	Mach	Mass	Altitude	Alpha	Flight path Gamma
Take-off Roll	0.185	413,949	0	13.54	0
1st Climb	0.185	413,699	0	11.63	0
2nd Climb	0.23	413,634	10.7	8.67	5.74
3rd Climb	0.82	413,582	122	7.24	4.86
Cruise	0.82	409,957	10059	3.25	0
1st Decent	0.46	265,998	10059	1.93	0
2nd Decent	0.23	268,224	457.2	1.99	-2.85
3rd Decent	0.23	265,683	15.24	6.5	6.0
Landing Roll	0	265,681	0	0	0

STABILITY & CONTROL

Table C-6 Stability and control result

Flight condition	Static margin	CG X position	Neutral Point	Trim delta
Take-off Roll	5.7%	29.71	31.09	-1.21
1st Climb	17.67%	29.71	33.91	-7.46
2nd Climb	12.38%	29.71	32.65	-0.93
3rd Climb	13.13%	29.61	32.72	2.37
Cruise	20.74%	27.26	32.19	-13.45
1st Decent	21.04%	27.25	32.24	-10.55
2nd Decent	17.37%	27.27	31.39	-7.86
3rd Decent	14.73%	27.27	30.77	-18.45
Landing Roll	14.92%	27.27	30.82	-6.89

Table C-7 Mass breakdown result

Item	Mass (kg)	X (m)	Y (m)	Z (m)
Propulsion System	34,295	27.09	0	1.55
Fuselage	67,927	24.67	0	0.30
Wing	36,333	35.92	0	0
Control Surface	2,981	28.60	0	0.26
Nose Landing Gear	2,139	4.42	0	-5.32
Main Landing Gear	15,961	34.49	5.84	-4.07
Passenger	46,065	24.03	0	0.30
Payload	16,650	22.10	0	-1.55
Systems	33,137	28.60	0	0.26
Operational Items	6,774	28.60	0	0.26
Fuel Tank Wing 0	876	31.43	11.58	0.01
Fuel Tank Wing 1	519	32.78	16.47	0.07
Fuel Tank Wing 2	441	41.39	27.46	0.67
Fuel Tank Fuselage	26	6.63	0	-0.90
Fuel	149,819	33.82	0	0.17

Table C-8 Approximate fuel burn

Mission phase	Approximate fuel burn (kg)
Take-off Roll	250.19
1st Climb	64.18
2nd Climb	52.48
3rd Climb	3624.9
Cruise	141732.6
1st Decent	2226.23
2nd Decent	315.21
3rd Decent	2.14
Landing Roll	0
Total fuel	148268.97

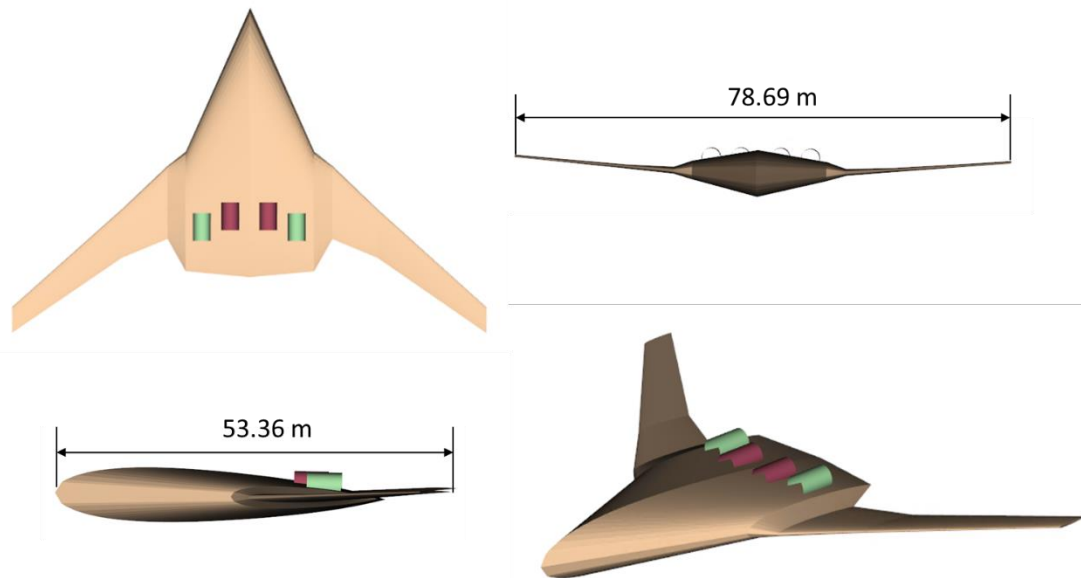


Figure C-9 General arrangement

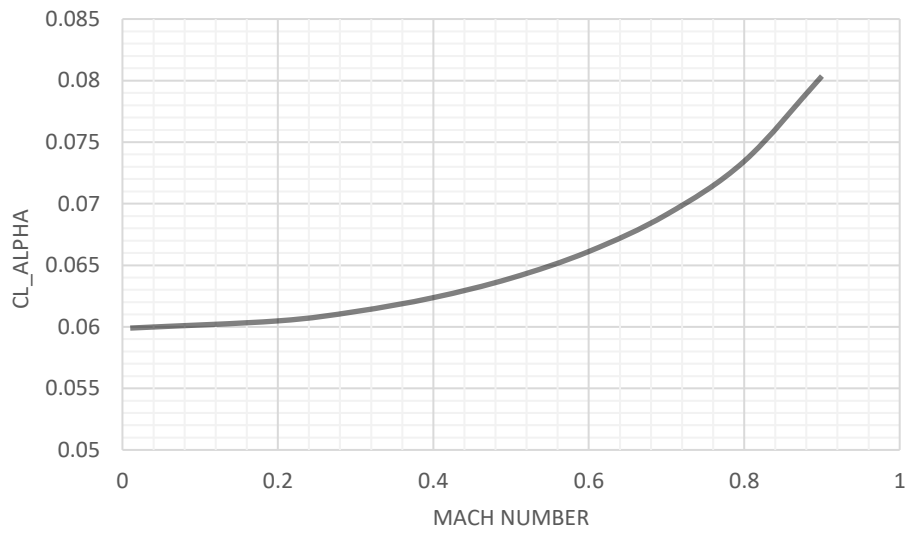


Figure C-10 Lift curve slope

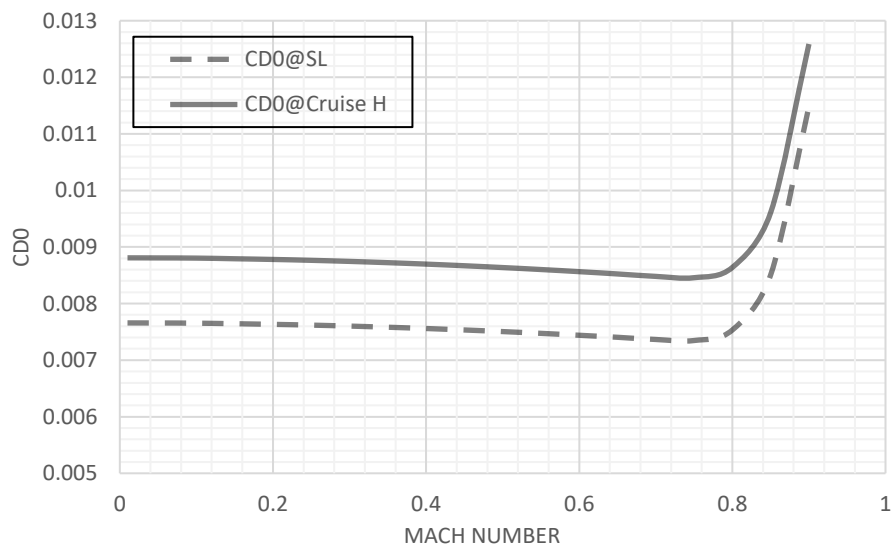


Figure C-11 Zero lift drag coefficient CD_0

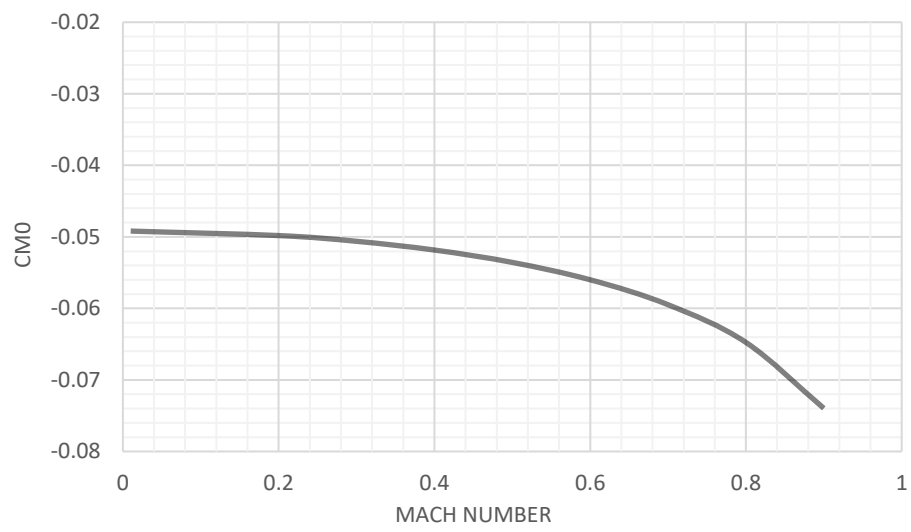


Figure C-12 Moment coefficient CM_0

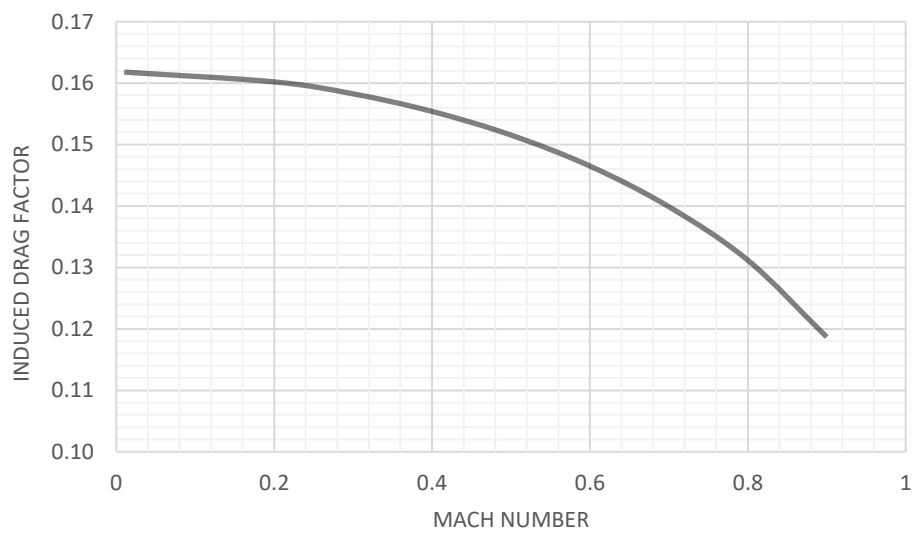


Figure C-13 Induce drag coefficient factor

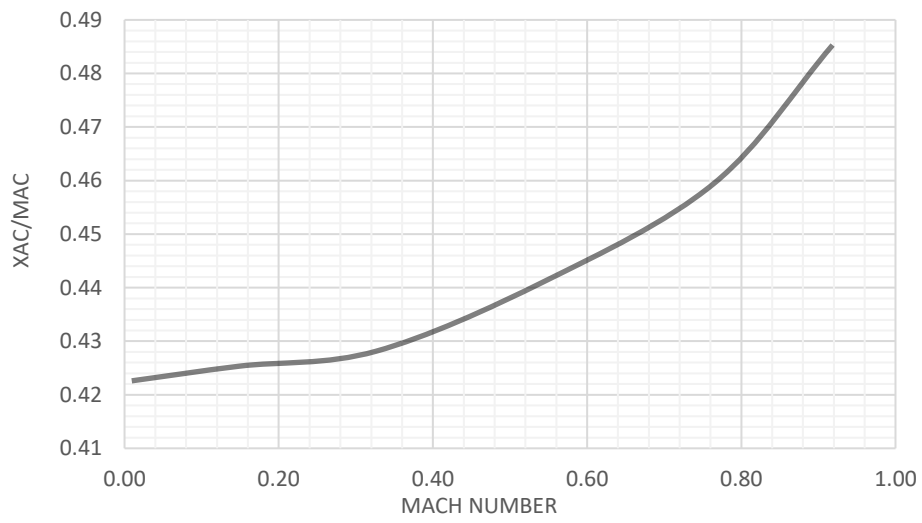


Figure C-14 Aerodynamic centre position

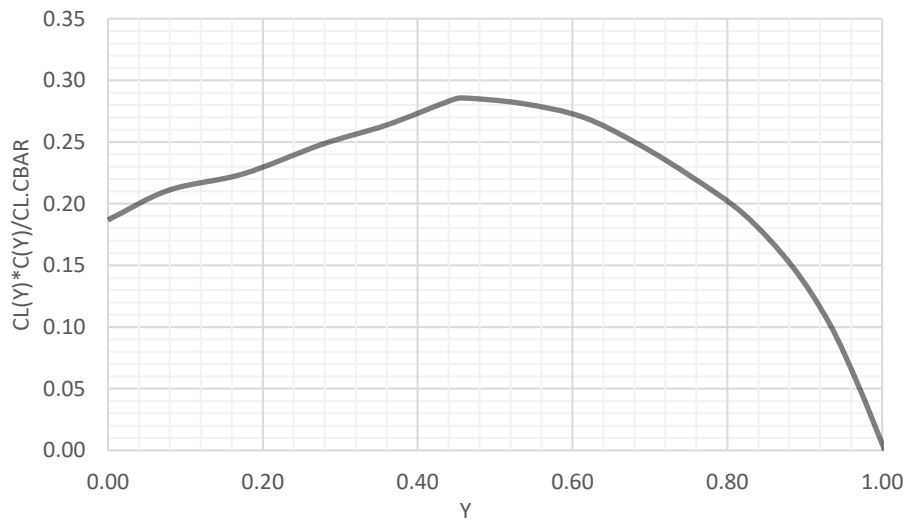


Figure C-15 Wing spanwise lift distribution

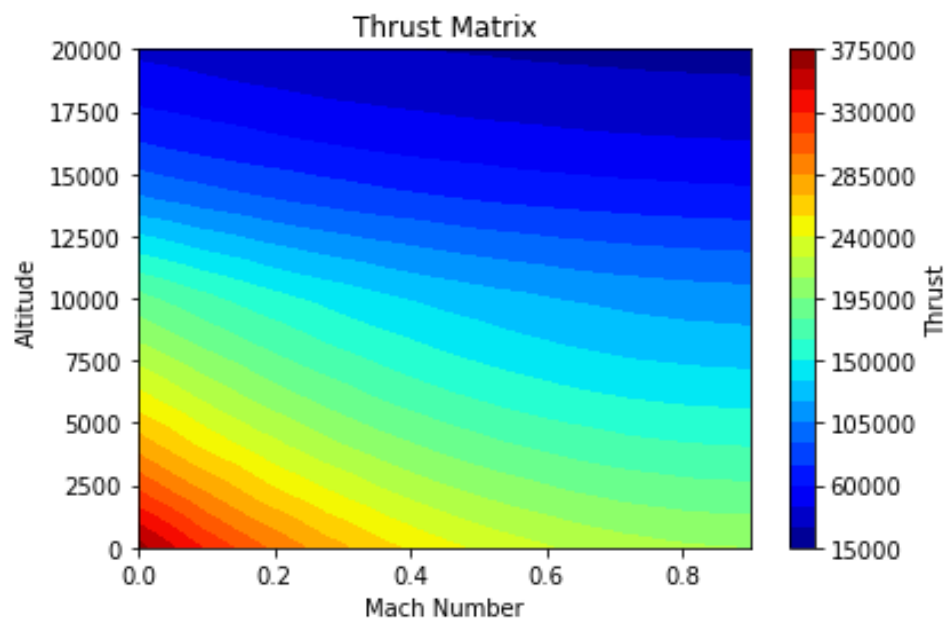


Figure C-16 Thrust map

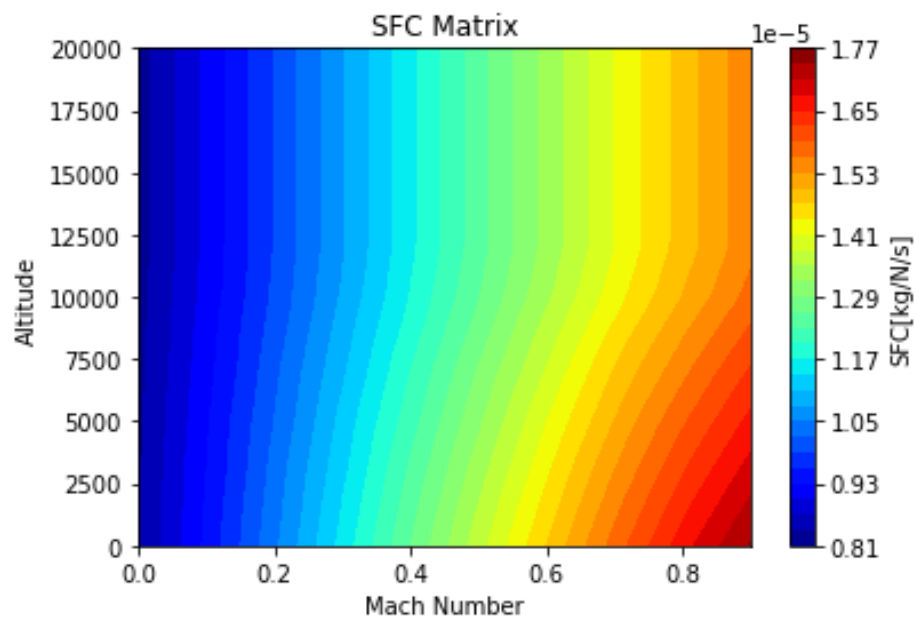


Figure C-17 SFC map

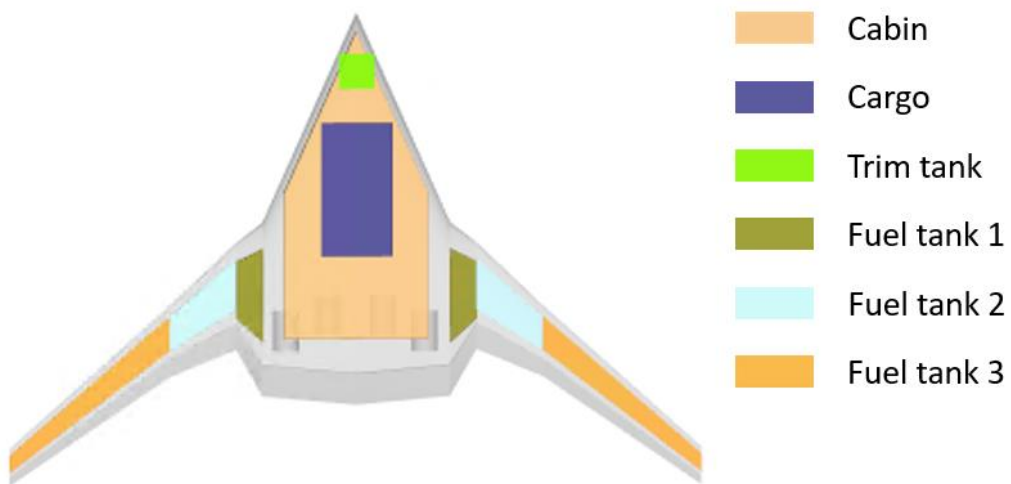


Figure C-18 Packaging layout