

CRANFIELD UNIVERSITY

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STRUCTURAL SIZING AND MASS ESTIMATION OF STRUT- AND
TRUSS-BRACED WINGS: THE EFFECTS OF MODERATE TO
HIGH ASPECT RATIOS AND NOVEL TECHNOLOGIES

SCHOOL OF AEROSPACE, TRANSPORT AND
MANUFACTURING
PhD AEROSPACE

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Academic Year: 2019 - 2023

Supervisor: Prof Howard Smith
Associate Supervisor: Prof Joseph Loughlan
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ABSTRACT

The aviation sector aims for more efficient and environmentally friendly novel transport aircraft. For this aim, sensitive and rapid conceptual or early preliminary design stage methods are vital to accurately analysing and comparing potential candidates. Thus, this PhD thesis presents a comprehensive, physics-based (quasi-analytical) structural sizing and mass estimation method for conventional and innovative aircraft wings. The method is unique due to its computational efficiency and ability to cover a wide spectrum of wings, ranging from moderate to ultra-high aspect ratios, accommodating new wing configurations such as strut- and truss-braced wings with composite or metallic materials, and incorporating emerging propulsion technologies such as hydrogen, electric and distributed propulsion systems.

A key distinguishing feature of this study is the exclusive focus on the load-carrying structures of an aircraft wing: the wing boxes, struts, juries, and offsets. Validation was an integral part of the model's development, tested against data from 14 aircraft: 13 real aircraft with aluminium wing boxes and an aircraft with a high aspect ratio truss-braced wing and a composite wing box. The results show the model's high accuracy and correlation with actual wing group mass data, featuring an average error of -2.22% and a standard error of 1.74%.

Simultaneously, the thesis offers comprehensive literature reviews and identifies significant gaps in the field. It provides thorough parametric and comparative studies to increase the credibility of the presented methods and investigate the effect of design decisions on the mass of the studied aircraft wings. Four hundred eighty-three loading cases are examined to detect the critical ones that drive the wing components' structural sizing. A reduced number of nine loading cases is introduced, reducing the computational time by thirtyfold. Furthermore, primary optimisation studies are conducted with the developed models for optimum strut-wing attachment and engine locations, considering different engine counts.

Keywords:

Strut-Braced Wing, Truss-Braced Wing, Structural Sizing, Mass Estimation, Weight Estimation, Aircraft Design, Distributed Propulsion, Hydrogen Aircraft, Electric Aircraft, Parametric Study, Comparative Studies, Wing Box, Critical Loading Cases.

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NOMENCLATURE

$WBF_{u,j}$, $WBF_{l,j}$	Coordinate points of the wing box outline in the y and z axes. The subscripts of u and l are for the upper and the lower surfaces, respectively.	mm
$WBT_{l,j}$, $WBT_{t,j}$	Coordinate points of the wing box outline in the x, y axes. The position is l for LE and t for the trailing edge. The subscripts of l and t are for the leading and trailing edges, respectively.	mm
$WF_{u,j}$, $WF_{l,j}$	Coordinate the wing planform's outline points in the y and z axes. The subscripts of u and l are for the upper and the lower surfaces, respectively.	mm
$WT_{l,j}$, $WT_{t,j}$	Coordinate the wing planform's outline points in the x, y axes. The subscripts of l and t are for the leading and trailing edges, respectively.	mm
η_{Tank}	Fuel tank efficiency factor	
$\overline{L}_\xi$	Rolling moment of the wing	Nmm
$\{M_j\}, \{\dot{M}_j\}$	List of moment components at the local axes and transformed axes, respectively.	Nmm
$\bar{Z}_{Aileron,case}$	The actual total lift generated by the aileron	N
$\bar{p}_{case}$	Angular rolling acceleration	rad/s^2
$\Delta M_{y,Cm,j}$	Moment increment caused by the wing pitching moment at the jth wing section	Nmm
$\Delta M_{y,Con,j}$	Moment increment caused by the concentrated loads at the jth wing section	Nmm
$\Delta M_{NO,1}$	Mass penalty due to the sheet taper and joints in the panels and cut-outs	kg
$\Delta M_{NO,2}$	Mounting and connection mass penalty	kg

ΔW_{ST}	Mass penalty caused by the aeroelastic constraints	kg
$A_{6-sided,j}, A_{6-sided,j+1}$	The cross-sectional area of the wing box at j th and $j + 1$ th sections, respectively	mm^2
AF_j	Distance between the leading edges of the wing sections in the z-axis	mm
AT_j	Distance between the leading edges of the wing sections in the x-axis.	mm
$C_{L,p,j}$	Updated local lift coefficient with aerodynamic damping at j th section	
$F_{Disturbing}$	Transverse load (caused by strut buckling) applied to the jury	N
$F_{Jury,i}$	Axial load acting on the jury	
$F_{Man,j}$	Loads generated in symmetric manoeuvring	N
$F_{Roll,j}$	Loads generated in rolling manoeuvre	N
$F_{Total\ Shear,j}$	Total loads, causing the shear load at the j th wing section	N
$F_{Total\ inertia,j}$	Total inertia load at the j th wing section.	N
I_{xx}, I_{yy}, I_{zz}	Mass or area moment of inertias about x, y , and z axes, respectively	Nm^4
L_ξ	ND rolling moment coefficient from specifications	
$M_{Bending\ Total}$	The total mass of the bending-resistant material	kg
M_{Caps}	The total mass of the spar caps	kg
$M_{Comp\ Box,j}$	The total mass of the j th section of the load-carrying boxes (strut, jury, or offset)	kg
$M_{Comp\ LE,j},$ $M_{Comp\ TE,j}$	Mass of the fixed leading or trailing edges at j th section, respectively, of the strut, jury, or offset	

$M_{Inertia,j}$	Total mass causing inertia loads at the jth wing section	kg
M_{Jury-1}	The total mass of Jury-1	kg
M_{Jury-2}	The total mass of Jury-2	kg
M_{Offset}	The total mass of the offset	kg
$M_{Rib,Howe}$	Estimated total rib mass using Howe's (1996) method	kg
$M_{Rib,Torenbeek}$	Estimated total rib mass using Torenbeek's (1992) method	kg
$M_{Shear Total}$	The total mass of the shear-resistant materials	kg
$M_{Shear,l}$	Mass of lower surface shear-resistant materials	kg
$M_{Shear,u}$	Mass of upper surface shear-resistant materials	kg
$M_{Stringers}$	The total mass of the stringers	kg
M_{Strut}	The total mass of the strut	kg
$M_{WB Other}$	Other mass components of the wing box	kg
$M_{WB Structure}$	Mass of the wing box load-carrying structure	kg
$M_{Wing Box}$	Total wing box mass	kg
$M_{Wing Group}$	Total wing group mass	kg
$M_{Wing Other}$	Other mass components of the wing group	kg
$M_{Wing fuel,j}$	Mass of the fuel stored inside the wing box	kg
M_x, M_y, M_z	Moments about x, y , and z axes, respectively	Nmm
$M_{y,Cm,j}$	The total moment caused by the wing pitching moment at the jth wing section	Nm
$M_{y,Con,j}$	The total moment is caused by the concentrated loads at the jth wing section	Nm

$N_{Boom,j}$	Total number of mass booms, including the spar caps	
N_{Case}	Total number of loading cases	
N_{Sec}	Total number of sections in a structural component	
$P_{Ail,in}, P_{Ail,out}, P_{Ail,j}$	Lift per unit length at the root, tip, and any point of the aileron in the aileron's spanwise direction	N/mm
$P_{Damping,j}$	Damping load per unit length along the wing	N/mm
$P_{Ellip,Root}, P_{Ellip,j}$	Load per unit length values at the root, and any position of an elliptical lift distribution, respectively	N/mm
$P_{Trap,Tip}, P_{Trap,Root}, P_{Trap,j}$	Load per unit length values at the tip, root, and any position of a trapezoidal lift distribution, respectively	N/mm
$P_{Tri,Tip}, P_{Tri,Root}, P_{Tri,j}$	Load per unit length values at the tip, root, and any position of a triangular lift distribution, respectively	N/mm
R_S	The reaction force applied by the strut to the wing	N
$R_x(\beta), R_y(\alpha), R_z(\theta), [R]$	Rotation matrixes for roll, pitch, and yaw, respectively and $[R]$ is the general transformation matrix	
$S_{Aileron}$	Aileron surface area	mm^2
SCR_j	The ratio of the cross-sectional area of the stringers to that of spar caps	
S_{Wing}	Wing area	mm^2
V_A, V_B, V_C, V_D	Design manoeuvring speed, design speed for maximum gust intensity, design cruising speed, and design dive speed, respectively	mm/s^2
$V_{Fuel\ tank,Max,j}$	The total volume of the internal space of a wing box section	mm^3
$V_{Fuel\ tank,j}$	The volume of the fuel tank inside the jth wing box section	mm^3

$V_{TAS,Case}$	True airspeed of the flight case	mm/s
V_x, V_y, V_z	Shear forces in the direction of x, y , and z axes	N
$W_{Wing\ fuel,Case,j}$	The weight of the fuel stored in the wing	N
$W_{Wing,Initial,Chosen}$	Initial wing mass estimation	N
$W_{Wing-bay,j}$	Weight of the wing bay	N
$Z_{\xi,HS}, Z_{\xi,LS}$	Lift generated by high and low-speed ailerons, respectively	N
$b_{Bay,j}$	The length of the jth wing bay	
$c_{Ellip,j}$	Chord of equivalent elliptic wing planform at the jth wing section	mm
$c_{Schrenk,j}$	Chord of equivalent Schrenk wing planform at the jth wing section	mm
c_j	Chord length at the jth wing section	mm
$c_{m,j}$	Moment coefficient of the jth wing section	
k	Bending stiffness	N/mm
l^*	Length of the strut between juries	mm
$l_{Circum. Length}$	Circumference of the circular fixed leading or trailing edges	mm
$l_{Ef,Comp}^*$	Effective length	mm
p_{case}	Angular rolling speed	rad/s
p_j	Distributed weight value at the jth wing section	N/mm
$t_{Web\ front,j}, t_{Web\ rear,j}$	The final thickness of the front and rear spar webs, respectively, at the jth wing section	mm

$t_{Buckling,j,i}$	The minimum thickness of the i th wing box panel at the j th wing section to resist the panel buckling with plasticity effect	mm
$t_{Comp,LE,Skin,BS,j}$, $t_{Comp,TE,Skin,BS,j}$	The minimum thickness of the fixed leading and trailing edges at j th section to resist the bird strike	mm
$t_{Comp,Skin,Bd,j}$	The minimum skin thickness of the load-carrying box at j th section to resist the combined bending and axial stresses	mm
$t_{Comp,Skin,Bk,j}$	The minimum skin thickness of the load-carrying box to resist the Euler column buckling at j th section	mm
$t_{Comp,Skin,LB,j}$	The minimum skin thickness of the load-carrying box at j th section to resist the local buckling	mm
$t_{Comp,Skin,Selected}$	The selected minimum skin thickness of the load-carrying box considering all the constraints and loading cases	mm
$t_{Comp,Skin,St,j}$	The minimum skin thickness of the load-carrying box at j th section to resist the yield or ultimate stresses	mm
$t_{Comp,box,j}$	The maximum thickness of the rectangular box between the upper and the lower surfaces	mm
$t_{E.Buckling,j,i}$	The minimum thickness of the wing box panel to resist the elastic panel buckling	mm
$t_{Panel\ section,u,j}$, $t_{Panel\ section,l,j}$	The minimum thicknesses of the upper and lower surface wing box panels, respectively, resist the shear flow and buckling under different loading cases	mm
$t_{Panel,j,i}$	Minimum wing box panel thickness to resist the shear flow and buckling	mm
$t_{Web\ rear,j}$, $t_{Web\ front,j}$	The thickness of the rear and front spar webs, respectively	mm

$t_{components\ name}$	The thickness of the structural element or component specified	mm
$t_{shear\ j,i}$	The thickness of the wing box panel to resist the shear flow	mm
v_{Strut}	The transverse deflection of the strut	mm
$w_{Comp,box,j}$	The maximum width of the load-carrying box between the leading and the trailing edges	mm
$w_{Panel,j,i}$	Panel width between the mass booms	mm
x_o, y_o	The coordinates of the box skin in the local x and y axes	mm
y_{HS}, y_{LS}	The moment arm of the HS and LS ailerons from their pressure centre to the aircraft centre line	mm
y_{ξ}	The equivalent moment arm length of the ailerons	mm
ν_{xy}	Poisson ratio	
τ_{Trap}	The ratio between the root and the tip values of $P_{Trap,j}$	
C	Concentrated load	N
A	Cross-sectional area	mm^2
B	The cross-sectional area of the mass booms	mm^2
E	Material elastic modulus	N
F	Force	N
K	Buckling coefficients obtained from references	
L	Lift load	N
M	Indicates mass or moment depending on the content	kg or Nmm
P	Any applied external load	N
T	T is torque and equal to M_y for the wing	Nmm

W	Weight or distributed load depending on the content	N or N/mm
b	Wingspan	mm
d	Denotes the distance between any two points in a geometric model or moment arms if stated.	mm
i, j, k	Indices in the x, y, and z axes, respectively	
l	Length	mm
n	Loading factor	
x, y, z	Axes of a coordinate system	mm
g	Gravitational acceleration	mm/s

Subscripts

$(\cdot)_{Aileron}$	Indicates that the parameter belongs to the aileron
$(\cdot)_{Air}$	Indicates that the parameter belongs to the air
$(\cdot)_{Aircraft}$	Indicates that the parameter belongs to the whole aircraft
$(\cdot)_{Arm}$	Moment arm
$(\cdot)_{Case}$	Shows that the value of the variable changes with the flight case
$(\cdot)_{Con}$	Indicates that the parameter belongs to a concentrated load
$(\cdot)_{Ellip}$	Indicates that the parameter belongs to an elliptical distribution
$(\cdot)_{Fuel}$	Indicates that the parameter belongs to the fuel
$(\cdot)_{HS}$	Indicates that the parameter belongs to a high-speed aileron
$(\cdot)_{LS}$	Indicates that the parameter belongs to the low-speed aileron
$(\cdot)_{Man}$	Indicates that the parameter belongs to a symmetric manoeuvring flight case.
$(\cdot)_{Root}, (\cdot)_{Tip}$	Shows the value of the parameter at the root and the tip of the component, respectively

$(\cdot)_{Schrenk}$	Indicates that the parameter belongs to Schrenk`s lift distribution method
$(\cdot)_{Trap}$	Indicates that the parameter belongs to a trapezoidal distribution
$(\cdot)_{Tri}$	Indicates that the parameter belongs to a triangular distribution
$(\cdot)_{Wing}$	Indicates that the parameter belongs to the whole wing
$(\cdot)_j$	j indicates the section number of the component along its $y - axis$
$(\cdot)_m$	Indicates that the unit of the parameter is a meter
$(\cdot)_{mm}$	Indicates that the unit of the parameter is a millimetre
$(\cdot)_\xi$	Indicates that the parameter is related to aileron deflection

Greek Symbols

α_1	Lift curve slope	
$q_{j,i}$	Shear flow at the jth wing section and ith panel	N/mm
Λ_j	Non-dimensional length as a fraction of the half-span length	
Ψ_j	ND length as a fraction of wing thickness at jth wing section	
α_j	Wing section incidence angle	<i>degree</i>
β_j	Anhedral of the wing box reference line	<i>degree</i>
ε_Y	Yield strain	
ε_j	ND chordwise length as a fraction of the local chord	
η_B	Plasticity factor for panel buckling	
θ_j	Leading edge sweep angle	<i>degree</i>
$\lambda_{Aileron,In}$	ND location of the aileron`s inboard section	
$\lambda_{Aileron,Out}$	ND location of the aileron`s outboard section	
ρ_{Air}	Air density	kg/mm^3
ρ_{Bend}	The density of the bending resistance material	kg/mm^3

ρ_{Fuel}	Fuel density	kg/mm^3
ρ_{Shear}	The density of the shear resistance material	kg/mm^3
$\sigma_{LocalBuck}$	The local buckling stress of the rectangular box	N/mm^2
$\sigma_{U,i}, \sigma_{Y,i}$	Direct ultimate and yield stresses, respectively	N/mm^2
$\sigma_{U,60\%,j,i}, \sigma_{Y,60\%,j,i}$	Shear ultimate and yield stresses, respectively	N/mm^2
$\tau_{cr,E}$	Critical elastic buckling of a panel	N/mm^2
τ_j	Thickness to chord ratio at jth wing section	
$\phi_{Strut,i}, \gamma_{Strut,i}$	The angular deflection of the strut due to column buckling at the fuselage and wing attachments, respectively	<i>degree</i>
Δ	Increment	
ξ	Aileron deflection	<i>degree</i>
σ	Direct stress	N/mm^2
τ	Denotes shear stress or ratio depending on the content and subscripts	

Acronyms

2D	Two Dimensional
3D	Three Dimensional
AR	Aspect Ratio
CG	Centre Of Gravity
CLT	Classical Laminate Theory
CS.23	Certification Specifications And Acceptable Means Of Compliance For Normal, Utility, Aerobatic, And Commuter Category Aeroplanes CS-23
CS.25	Certification Specifications And Acceptable Means Of Compliance For Large Aeroplanes CS-25

EAS	Equivalent Air Speed
EASA	European Union Aviation Safety Agency
Eq. Eqs.	Equation, Equations
HARW	High Aspect Ratio Wing
LE	Leading Edge
ME	Mass Estimation
MMoI	Mass Moment Of Inertia
ND	Non-Dimensional
OAT	One-Factor-At-A-Time
OEW	Operating Empty Weight
SBW	Strut-Braced Wing
SDaME	Structural Design And Mass Estimation
SSaME	Structural Sizing And Mass Estimation
TAS	True Air Speed
TBW	Truss-Braced Wing
TE	Trailing-Edge
TLAR	Top-Level Aircraft Requirements
TOGW	Take Off Gross Weight
UHARW	Ultra-High Aspect Ratio Wing
WB	Wing Box
WE	Weight Estimation
MM	Mathematical Model
DE	Differential Evolution

SLSQP

Sequential Least Squares Programming

1 INTRODUCTION

Since the advent of the 20th century, air vehicles have served humanity in several vital areas. They stand as the most rapid mode of transportation known to humankind and have been improved remarkably since the first successful powered flight, pioneered by Wright Brothers, on December 17, 1903.

Over the preceding decades, the domination of the cantilever wing and tube concept has been irrefutable in civil aviation. Although some innovative aircraft configurations have been considered, they have yet to be utilised in civilian transportation. However, the pursuit of developing a more advantageous and competitive aircraft configuration aligned with sustainable and environmentally friendly development goals has remained steadfast in ongoing research efforts.

In this regard, the future trajectory of aviation is likely to be shaped by the innovative configurations examined within this study's scope and those explored in the broader academic literature. Therefore, the principal objective of this thesis is to proffer novel aircraft design methods which have the potential to transform aspirations into tangible reality.

1.1 Aims of Aviation

The aviation sector progresses yearly with favourable economic resources and adverse environmental impacts. The total number of flights grew by 8% from 2014 to 2017 and is expected to increase by 42% from 2017 to 2040, according to the reports presented by European Environment Agency et al. (2019). Future aviation targets are proposed by academia, commercial organisations or state agencies to alleviate the unfavourable effects of the rapidly growing aviation sector (ACARE, 2015; European Environment Agency, European Aviation Safety Agency and Eurocontrol, 2016, 2019; Boeing Commercial Aviation, 2022).

Previously Darecki & Edelstenne (2011) presented the Flight-path 2050 report of the European Commission with the future goals of aviation in cooperation with Air Transport Action Group ATAG (2012). Some targets presented in the report are to reduce the CO_2 by 75% per passenger kilometre, NO_x by 90% and perceived noise by 60% compared to the conventional aircraft in 2000. In order to achieve these goals, many researches have been focused on air traffic management & operations, airports, future propulsion

systems, fuels and novel aircraft configurations (European Environment Agency, European Aviation Safety Agency and Eurocontrol, 2016, 2019).

There are different disciplines in designing and manufacturing an air vehicle, and each has been researched to meet the proposed targets. Contribution from each sub-discipline and organising their collaborations in a multidisciplinary design optimization (MDO) environment are essential to achieve flagship projects. Specialists from different disciplines investigate novel technologies and aircraft concepts to achieve sustainable, competitive, environment-friendly products.

1.2 Possible Solutions to Meet the Targets

There have been several types of research focused on reducing the fuel consumption of aircraft by optimising mission specifications (Variyar et al., 2014), using alternative propulsion technologies such as counter-rotating open rotor (CROR) engines (Kennedy et al., 2013; Plythe & Smith, 1985; Moerland et al., 2017), utilising alternative fuels (Hendricks, Bushnell and Shouse, 2011; Hoelzen *et al.*, 2018; Nicolay *et al.*, 2021), employing natural laminar flow method for wing (Ishikawa *et al.*, 2012; Ma, Abouhamzeh and Elham, 2023) and nacelle (Hastings et al.1986), and studying boundary-layer ingestion inlets (Rodriguez, 2009) and other technologies—all these approaches aimed to reduce fuel consumption by modifying conventional aircraft layouts. However, combining these methods with novel aircraft configurations can reduce fuel consumption further (Moerland *et al.*, 2017).

The goals seem unachievable with the conventional aircraft configurations comprising a tube and a wing (Rajpal et al., 2018; Hari, Schetz and Kapania, 2019). Novel aircraft concepts such as blended-wing body (Liebeck, 2004; Liu, Karpuk and Elham, 2020) and joined-wing (Wolkovitch, 1985) have been demonstrated over the last decades with promising reductions in fuel consumption. Another candidate to provide solutions to the issues mentioned above is reported to be a strut-braced wing (SBW) or truss-braced wing (TBW) aircraft concept, which is grouped under the joined-wing concept by Cavallaro and Demasi (2016).

1.3 Why Strut- or Truss-braced Wings?

In its Subsonic Ultra Green Aircraft Research N+3 (SUGAR N+3), NASA investigated the performances of different types of novel aircraft concepts, including conventional, blended wing body and SBW, to find a solution for the requirements mentioned above,

including environmental concerns and aircraft performances. An aircraft design scenario for a medium size aircraft with 154 passengers for a 900 *NM* target was created, and the performance of each aircraft configuration was examined. The SBW configuration provided the best solution among all the candidates regarding fuel efficiency (Bradley and Droney, 2011).

In another research under the WeCare project of the German Aerospace Centre (DLR), Linke et al. (2016) presented that SBW configuration combined with CROR engine can reduce the fuel consumption of a 100-150 seat category aircraft by 26% in 2050 and 45% of NO_x reduction can also be achieved with new combustion technologies. The result from the environmental effect evaluation showed that SBW configuration could reduce the global warming caused by aircraft of that type by 17%, considering the expected progress in the future.

Another advantage of SBW is its similarity to conventional aircraft, about which we have a wide range of knowledge compared to other novel configurations. Hence several components of SBW aircraft can be designed and analysed with most existing design methods relatively easily compared to different novel aircraft configurations. It is also likely that SBW aircraft have fewer uncertainties in their performance analysis due to their similarity to conventional aircraft. Regarding the risk taken for a novel configuration by airliner companies, SBW might also be a favourable option.

In addition to the findings presented in this section, a comprehensive review of the SBW and TBW studies will be presented in Chapter - 2 (LITERATURE REVIEW).

1.4 Terminology for Strut- or Truss-braced Wings

Before proceeding further, explaining the terminology used for SBW (as illustrated in Figure 1) and TBW configurations would prevent confusion. This section briefly describes the terminology used in this work for SBW and TBW configurations following the references (Frank H. *et al.*, 2001; Cavallaro and Demasi, 2016).

Figure 2 illustrates the main components of the cantilever, strut-braced wing (SBW), 1-jury truss-braced wing (1TBW) and 2-jury truss-braced wing (2TBW). In SBW and TBW configurations, the strut, jury and offset are only structural components, and they are typically not used as lift-generating surfaces. However, they are prone to aerodynamic loads. The difference between the TBW and SBW configurations is the addition of the jury, and those configurations can be designed with or without offset. The SBW can

sometimes be used as a general term in this thesis to describe the SBW and TBW concepts unless their differences are emphasized.

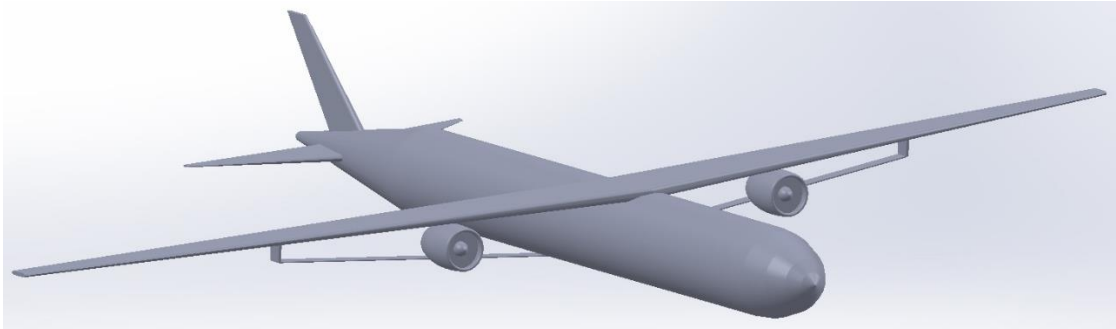


Figure 1 A 3D view of an SBW aircraft with an offset generated in SolidWorks. The geometric parameters were derived from the geometric models of this study.

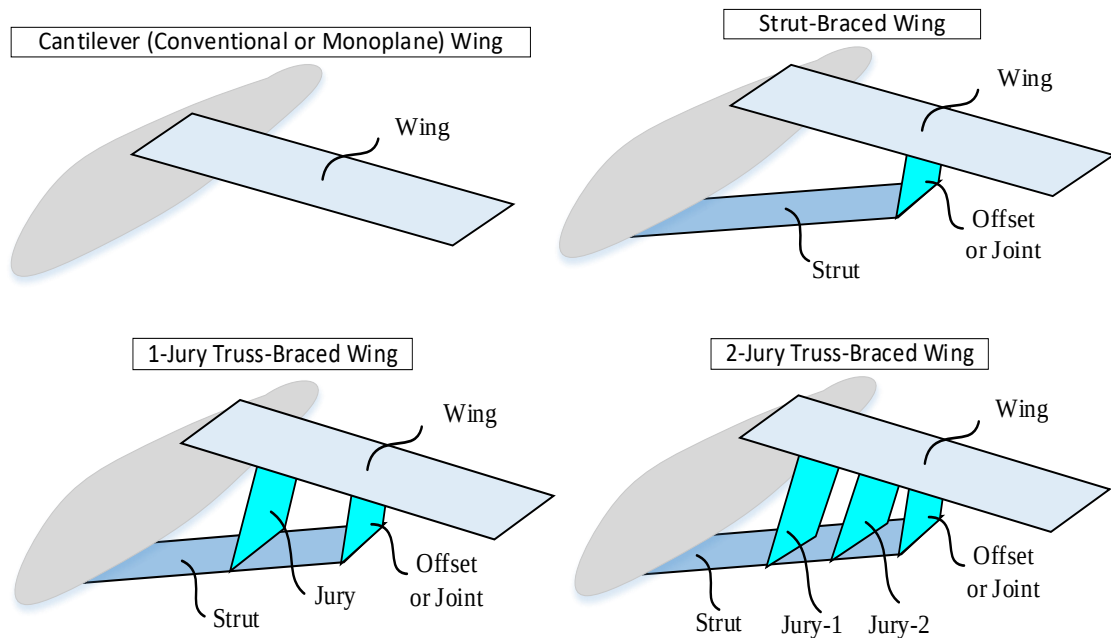


Figure 2 Terminology used for the strut- and truss-braced wings.

1.5 Advantages of Strut-braced Wings

SBW is one of the first concepts that has been used since the early years of aviation; however, due to its high profile drag penalty caused by the strut compared to the cantilever wing, it has not gained extensive popularity in aviation except small low-speed aircraft.

Conversely, using a strut enables the use of high aspect ratio wings (HARW) with less wing group mass penalty compared to its cantilever equivalent (Potter, 2017), and this

is illustrated in Figure 3 with a graph generated using the methods developed in this thesis. Increasing the wings' aspect ratio (AR) reduces the induced drag with less additional mass in TBW than cantilever wings. Induced drag was reported to account for 80%-90% of the total drag at low flight regimes and around 30%-40% at cruise speed (Smith, 1996). Reduced induced drag combined with novel technologies can further increase the benefits of SBW and TBW aircraft (Gur et al., 2010) to achieve the above-presented rudiments of the European Commission.

With SBW, smaller wing thickness is also possible, which reduces the transonic wave drag hence the wings with lower sweep angles become possible. High aspect ratio (HAR), lower sweep angle and low airfoil thickness result in natural laminar flow due to a low Reynolds number. Therefore a remarkable increase in aircraft performance is obtained with SBW and TBW (Joslin, 1998).

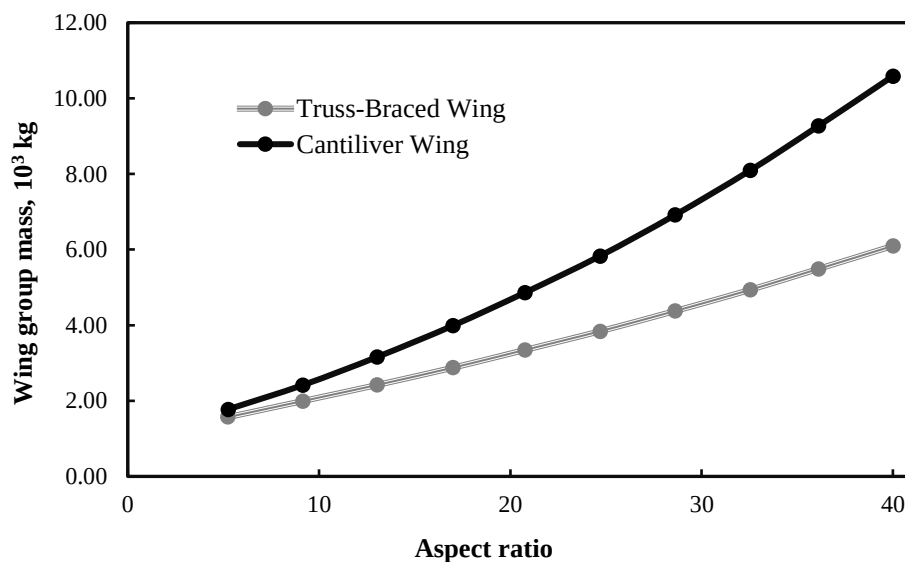


Figure 3 Wing weight variation of TBW (details given in Table 57) and its conventional wing counterpart with increasing aspect ratio.

1.6 General Gaps in the Literature of Strut-braced Wings

Although the design studies of the SBW configurations are expected to be more feasible compared to other novel configurations, there is still undiscovered design space for SBW aircraft configuration. New design methods covering mass estimation (ME), structural, aeroelastic and flight dynamic analyses need to be developed to optimise the SBW configuration further as a future competitor of air transport. Interface drag, for instance, between strut-wing and strut-fuselage joints is a local issue which has already been

tackled with different proposed solutions (Loma and Beach, 2018), and other solutions can be proposed.

A prosperous SBW design depends on the trade-off between lift-induced drag, interface drag, structural weight, and other constraints (Cheung, Rezgui, Cooper, & Wilson, 2019; Secco & Martins, 2018; Martins, Kennedy, & Kenway, 2014). However, the ME of the SBW is a significant and challenging task as there is little or no data for this configuration (Potter, 2017).

It will be discussed in Chapter - 2 (LITERATURE REVIEW) that there has been limited study in the literature for the structural design and mass estimation (SDaME) of the moderate to ultra-high AR cantilever wing, SBW or TBW transport aircraft configurations, exploiting novel propulsion technologies, including electric, hydrogen or distributed propulsions. The SDaME of these configurations and technologies with existing empirical or semi-empirical methods are expected not to provide accurate and sensitive enough results to be used in the conceptual or preliminary design exploration and optimisation studies to reveal the potential of the configurations.

Consequently, in this thesis, a mathematical model (MM) will be developed to respond to the research questions raised in the following section for the aircraft configurations mentioned above, considering the limits in their existing wing ME methods.

1.7 Research Questions

The main research question:

- How to analyse and estimate the structural design and mass of cantilever wings, SBWs, and TBWs, with moderate to ultra-high ARs equipped with distributed, hydrogen, electric, or conventional propulsion technologies at the conceptual or early preliminary design stage?

This question is discussed and replied to in Chapter- 2 (LITERATURE REVIEW), Chapter -3 (AIRCRAFT GEOMETRIC MODEL), Chapter - 4 (LOADING ACTIONS), Chapter - 5 (STRUCTURAL ANALYSIS AND SIZING), and Chapter - 6 (MASS ESTIMATION).

Subsequently, the following sub-questions are also addressed in the relevant chapters of the thesis.

1. What is the relationship between the AR and wing group mass in cantilever wings, SBWs, and TBWs? How accurately do empirical and semi-empirical methods capture this relationship? These questions are responded to in Chapter - 6 (MASS ESTIMATION).
2. What methodologies can be developed to estimate the mass of various components of the wing group, including the wing box, strut, jury, and offset? This question is discussed and replied to in Chapter- 2 (LITERATURE REVIEW), Chapter -3 (AIRCRAFT GEOMETRIC MODEL), Chapter - 4 (LOADING ACTIONS), Chapter - 5 (STRUCTURAL ANALYSIS AND SIZING), and Chapter - 6 (MASS ESTIMATION).
3. How do the design parameters of the wings and truss components, including strut, jury, and offset, influence the mass of those components? This question is addressed in Chapter - 7 (PARAMETRIC and SENSITIVITY STUDIES).
4. How does the choice of propulsion technology (distributed, hydrogen, or electric) impact the mass of cantilever and SBWs with moderate to high ARs? This question is addressed in Chapter 7 (PARAMETRIC and SENSITIVITY STUDIES) and Chapter 9 (COMPARATIVE and OPTIMISATION STUDIES).
5. Which critical loading cases drive the structural sizing of these wing configurations? This question is answered in Chapter - 8 (CRITICAL LOADING CASES INVESTIGATION).
6. How do material choices (composite or metallic) influence the wing mass of the studied aircraft configurations? The question is answered in Chapter - 9 (COMPARATIVE and OPTIMISATION STUDIES).
7. What strategies can be employed for the optimization of the studied configurations? This question is discussed in Chapter - 9 (COMPARATIVE and OPTIMISATION STUDIES).

The next section defines and presents the purpose and goals of this PhD thesis while considering the research questions.

1.8 Aim and Objectives

Aim: This thesis aims to develop conceptual or early preliminary design stage wing structural sizing and mass estimation methodologies with carefully chosen sensitivity and fidelity levels to study the design space of the novel, moderate to HAR cantilever wing, SBW and TBW with distributed, electric, or hydrogen propulsion technologies.

Objectives:

1. **Literature Review:** Conduct a comprehensive review of related literature to provide the basis for the novel research.
2. **Model Development:** Create a physics-based ME model for aircraft components (wing boxes, struts, juries, offsets), adaptable for different wing types and materials.
3. **Technological Integration:** Incorporate effects of emerging propulsion technologies (hydrogen, distributed electric systems) into the model.
4. **Analytical Solutions:** Develop efficient analytical solutions for calculating strut reaction force in the indeterminate SBW structure under varied loads.
5. **Model Validation:** Validate the model using real-world data from 14 aircraft, ensuring practicality and reliability.
6. **Result Analysis:** Evaluate the model's accuracy, aiming for a specific error range, to understand its precision and identify areas of improvement.
7. **Future Research Avenues:** Identify the current model's limitations and suggest improvement areas, such as in-depth composite design studies.
8. **Sustainable Aircraft Design:** Contribute to sustainable aircraft design by providing useful methods for designing efficient and environmentally friendly aircraft configurations.

1.9 Aircraft Design Stages

The aircraft design process comprises three main stages conceptual, preliminary, and detailed designs, and they are extensively discussed in several aircraft design books (Roskam, 1985; Howe, 2000; Torenbeek, 2013; Raymer, 2018).

In the initial phase, known as conceptual design, the aircraft's overall dimensions, shape, weight, and performance characteristics are determined, resulting in a basic layout. This stage primarily relies on simplified methods and models that require minimal input parameters, making them suitable for this specific design phase.

The focus shifts to structural and control system analysis, detailed wind tunnel testing and computational fluid dynamics calculations in the preliminary design stage. The core aircraft concept remains unchanged during this stage, with only minor adjustments.

Lastly, the detailed design phase represents the final stage, where precise designs for each aircraft component are developed for production. This stage includes determining

the size, quantity, and placement of structural elements and fasteners and designing manufacturing tools.

The three phases of aircraft design—conceptual, preliminary, and detailed—involve iteratively refining the aircraft's configuration and design using different models and analyses while ensuring compatibility and manufacturability.

1.10 What is a Model?

A system's model is a mathematical construct or representation that may foretell the system's performance under certain predetermined operational circumstances.

It is difficult to predict the effect of the design decisions on the outcome of the aircraft systems. Testing an aircraft system with the actual product is not sustainable in terms of resources and budgets. Therefore, models are usually used to predict better the system performance, which outcomes from specified design variables (Law, 2015). Nevertheless, it should also be emphasized that performance data of any actual operating product can also be used as an experimental result for future system designs.

Yet, in addition to the underlying systems' uncertainty, models occasionally can mislead modellers and designers (Savoie and Frey, 2012). For instance, a model may be compromised using inaccurate equations or false assumptions about a system's behaviour. Also, it's critical to understand the model's approach and acknowledge that it might not align with the designer's preferred strategy while dealing with other researchers' models. Therefore, extensive validation and verification studies are presented in this thesis to examine the capabilities and accuracy of the developed MM to prevent any uncertainties in its future use.

Models can be classified into two main groups; physical and mathematical, as depicted in Figure 4. Physical models, for instance, are utilised to test the behaviour of aircraft systems with small-scale representative models in wind tunnels. Besides, the models can also be classified as MMs, which can be either analytical or simulation-based (Law, 2015).

Both the physical and MMs are used in aircraft system design. But most of the aircraft models are mathematical, and they represent the aircraft system with reasonable and quantitative relations which can be manipulated to test the performance of the aircraft system or how it reacts to the design variables.

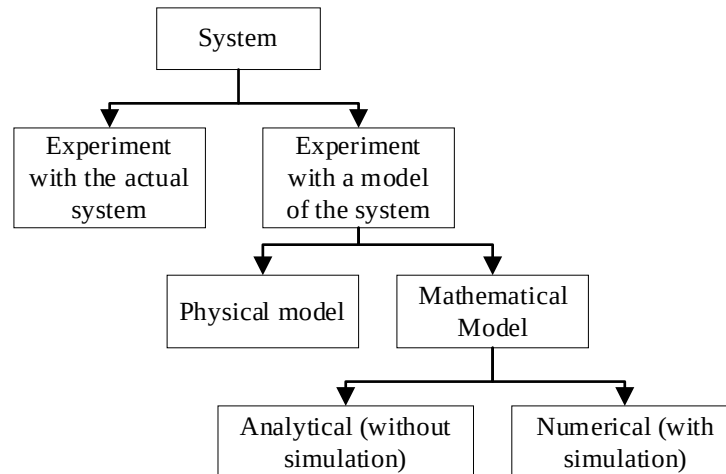


Figure 4 System design and analysis options (Law, 2015).

Suppose an analytical solution is available for an MM. In that case, using them instead of numerical simulations is desirable, which generally require higher computational power, computational time and labour work than analytical solutions (Flanagan, 2012; Law, 2015). It will be discussed in this thesis that although pure analytical models are not always available for aircraft systems, some simplifications can be applied to their use.

Mathematical models can be developed using empirical or physics-based approaches. The physics-based models are based on the governing equations that accurately represent the system's dynamics. On the other hand, the empirical models are generated by the available empirical data of a system using regression analysis. Models can also benefit physics-based and empirical approaches (Flanagan, 2012; Law, 2015), as preferred in this thesis.

1.11 Why are New Aircraft Design Methods Needed?

“If you do the right thing in the wrong phase, it is the wrong thing”. “Those who do not understand this always want to jump to detailed design too early; ensuring disaster” (Raymer, 2018).

In developing this PhD thesis, the wise words of Raymer (2018) should be noted. The balance between accuracy and complexity is critical in the initial design stages. Although they might lack precision, simple design methods are preferred for their ease of use and reduced design process complexity (as detailed in Section - 1.12 and Figure 6).

However, validating these simple methods becomes vital when dealing with novel aircraft concepts. As they are typically empirical and reliant on historical data, their efficacy can be challenged in the context of innovative designs. The soundness of this assumption will be further examined in Chapter - 6 (MASS ESTIMATION and VALIDATION).

Moreover, emerging technologies accompanying new aircraft layouts may further complicate these existing methods. This complexity is because these methods were developed in an era where such technologies were non-existent. Therefore, it is essential to recalibrate and validate these methods before employing them to design new concepts.

Nonetheless, there may be instances where even recalibrated methods fall short of meeting the design exploration objectives. Resorting to high-fidelity methods, though seemingly more accurate, can become highly impractical for design exploration studies. This approach can lead to premature detailed design, causing potential pitfalls, as warned by Raymer (2018).

In such scenarios, when sufficient historical data is lacking for new empirical equations and high-fidelity methods are impractical, the need arises for intermediate-fidelity level design methods. This thesis thus seeks to develop a comprehensive intermediate-fidelity MM for SDaME of SBW and TBW with novel technologies, filling a critical gap in the literature.



Figure 5 The relations between the literature review, gaps and mathematical model development

The proposed MM is designed to address the identified gaps in the literature, adhering to scientific requirements (as illustrated in Figure 5). After thoroughly reviewing existing literature on SBW and TBW aircraft, the gaps will be identified and an MM will be developed to address them.

1.12 The Challenge in the Model Development

When an MM is developed, the main challenge arises from the trade-off between its fidelity level and the level of aimed multidisciplinary design optimisation (MDO) studies, as illustrated in Figure 6.

The MM developed in this thesis is aimed to have an intermediate fidelity level as the development of the empirical methods is not feasible due to the limited historical data of the studied aircraft concepts.

The new model is also aimed to be used from trade studies to full MDO studies, as highlighted with the dashed line in Figure 6. Moreover, the low computational time of the developed model is also aimed for its use in MDO studies.

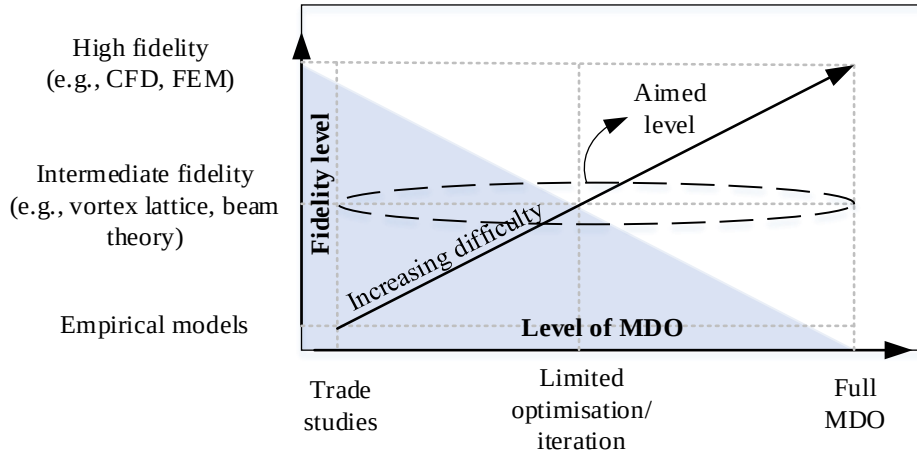


Figure 6 The trade-off between the fidelity level and the level of design optimisation studies (Agte *et al.*, 2010; Flanagan, 2012).

Following the features of the models discussed in this and the former section, the method developed in this thesis can be classified as an intermediate fidelity level physic-based MM, and it will further be discussed in Chapter - 2 (LITERATURE REVIEW). In the following chapters, the development of the new physic-based MM will be explained in detail.

1.13 Content of the Thesis

The following chapter (Chapter - 2 LITERATURE REVIEW) reviews the literature for the studied SBW and TBW aircraft and provides the required background for the subsequent chapters. All the disciplines used in the MDO of the studied concepts are reviewed, while special attention is given to the ME studies. As a result of the critical literature review, gaps are identified, and the new MM is designed to study those gaps.

Chapters - 3, 4, and 5 explain the sub-components of the new MM as illustrated in Figure 7. In Chapter - 3 (AIRCRAFT GEOMETRIC MODEL), new geometric models and relations are developed to represent the studied aircraft configurations with the required

fidelity level. The fidelity level of the geometric models is compatible with the loading action methods introduced in Chapter - 4 (LOADING ACTIONS) and structural analysis and sizing methods presented in Chapter - 5 (STRUCTURAL ANALYSIS AND SIZING). Chapter - 4 provides methods to compute the loads driving the sizing of aircraft structural components, and Chapter - 5 introduces the new methods for the stress analysis and sizing of the structural components of the novel wings.



Figure 7 Architecture of the new mathematical model for the novel wings' structural design and mass estimation.

The final ME formulas are given and grouped in Chapter - 6 (MASS ESTIMATION and VALIDATION), and their extensive validation studies are also presented in this chapter.

In addition to the validation studies, detailed parametric studies are provided in Chapter - 7 (PARAMETRIC and SENSITIVITY STUDIES) to examine the influence of the aircraft design parameters on the mass of the structural components of the studied novel wings.

In Chapter - 8 (CRITICAL LOADING CASES INVESTIGATION), the critical loading cases driving the sizing of the structural components are studied and reported. Finally, in Chapter 9 (COMPARATIVE and OPTIMISATION STUDIES), the novel and conventional configurations are compared to examine the effect of the new technologies. Moreover, optimisation studies are provided to show the capabilities of the new MM while investigating the optimums of the studied aircraft wing configurations. The following section introduces the details of each chapter and their contribution to the literature.

1.14 Novelties in this Study

This study introduces several innovative approaches to developing the MM and new findings from design and trade studies. These innovations can be summarised as follows and will be elaborated in greater detail in subsequent chapters:

Novelties in geometric models (Chapter - 3):

1. Novel geometric models have been developed and introduced for the fuselage, wing, strut, offset, and juries. These geometric relationships are initially verified using visualization tools and validated through ME results.

2. Unique load-carrying box models have been created for the wing, strut, offset, and juries. A high-fidelity hexagonal model is employed for wing box stress analysis, while a simpler rectangular box model is utilized for strut, jury, and offset stress analysis.

Novelties in loading actions (Chapter - 4):

1. The loading action module approach is innovative and tailored to accommodate various novel aircraft configurations.
2. A streamlined mass moment of inertia model is developed, offering rapid computation time and sufficient accuracy to meet the project's requirements.
3. Simplified methods for calculating aerodynamic loads driving the sizing of truss components are devised and presented.

Novelties in structural analysis and sizing (Chapter - 5):

1. The idealized hexagonal wing box model is modified, and its components' stress analysis and sizing in a non-iterative, direct manner are achieved using new equations derived in this thesis.
2. A comprehensive stress analysis and sizing procedure for the strut, jury-1, jury-2, and offset are developed, considering Euler column buckling, strength criteria, bending resistance, and local buckling.
3. While applying Castigliano's theorem for calculating strut reaction forces in indeterminate strut- or TBW configurations has been utilized in previous studies at Cranfield University, the solutions were limited to a single concentrated load. Concentrated loads caused discontinuities, necessitating a new analytical solution for each number of concentrated loads to compute strut reaction forces. In the literature, simple analytical approaches are restricted to one concentrated load, hindering the conceptual design of distributed propulsion SBW or TBW configurations. Therefore, for the first time in the literature, the author derived a general solution to Castigliano's theorem for an unlimited number of concentrated loads using a simplified TBW model.
4. One of the initial analytical methods is introduced to determine axial loads resulting from strut bending and acting on the juries. This conservative approach is sufficient to demonstrate that most designs' primary driving loads for jury sizing are mainly aerodynamic loads and bird strikes.
5. A novel, straightforward, and conservative model is also developed to size juries for bending loads caused by aerodynamic loads.

6. The analytical sizing procedure is provided for two jury concepts. First, the jury is designed normal to the wing reference line, and second, the jury is designed normal to the strut. Consequently, both configurations can be evaluated and compared. This procedure also sheds light on the behaviour of these configurations.
7. Following the macro mechanics of composites, a module for composite design and analysis has been developed. Although not extensively used in this study, the module is validated with other laminate analysis software. Quasi-isotropic materials are employed at this project stage, while the laminate design and analysis tool can be used for a more detailed composite structure design. The module also features an optimization capability for determining the minimum thickness of the composite laminate to withstand applied forces without laminate failure.

Novelties in mass estimation (Chapter - 6):

1. New ME formulas have been devised for load-carrying structures, such as the wing, strut, jury, and offset.
2. Two sizing approaches are developed for wing box components and compared, with further explanation in the validation studies.
3. Following multiple literature sources, a detailed categorisation of all mass groups is created and labelled with fidelity levels using a detailed aircraft group mass statement. This approach clarifies the entire thesis and can be utilised in future studies.

Novelties in parametric and comparative studies (Chapter - 7):

1. This thesis presents one of the most comprehensive parametric studies of SBW and TBW in the literature, which also serves to verify the developed MM.
2. These studies reveal the impact of design parameters on wing group and box mass.
3. Important design parameters affecting the wing group mass are identified for TBW aircraft with distributed, electric or hydrogen propulsion technologies.
4. The effect of the offset on wing group weight is presented for the first time in the literature by employing one of the first preliminary stage structural sizing methods for offsets. Contrary to previous statements, it is shown that an optimum offset length can reduce wing group weight while avoiding interface drags at strut and wing attachments.

Novelties in the critical loading case analysis (Chapter - 8):

- There has been no explicit outline in the literature for determining critical loading cases for conceptual or early preliminary structural design and sizing of TBW with distributed propulsion, hydrogen- or electric-powered technologies, and their combinations. From the analysis of approximately 2000 samples, nine critical loading cases are presented for the first time, ensuring no loss of accuracy in the MM. This approach significantly reduces computational time nearly 30-fold and can be adopted by other researchers.

Novelties in distributed propulsion and optimization (Chapter - 9):

1. The impact of distributed propulsion on TBW wing box weight is presented for the first time in the literature.
2. Additionally, global optimization algorithms are employed to optimize the location and engine counts in a distributed propulsion concept to reduce the wing group mass of the TBW concept even more.

2 LITERATURE REVIEW

In 1675, Isaac Newton made his most famous statement: “If I have seen further, it is by standing on the shoulders of Giants” (Chen, 2003).

Drawing inspiration from Isaac Newton's famous quote, this doctoral project aims to advance knowledge of moderate to HAR cantilever, strut- or truss-braced wing aircraft configurations with emerging technologies. Despite benefiting from the insights presented in the previous chapter, it remains crucial to conduct a more exhaustive review of the existing literature to identify gaps and propose meaningful contributions to the current body of knowledge.

Figure 8 visually represents the project's development and the systematic literature review process. This methodology aims to thoroughly assess past design studies on strut- or truss-braced wing aircraft as available in scholarly texts. In addition to examining individual disciplinary studies, multidisciplinary works have been also reviewed and discussed. These studies were categorized based on their relevant disciplines, and the gaps in each discipline were noted.

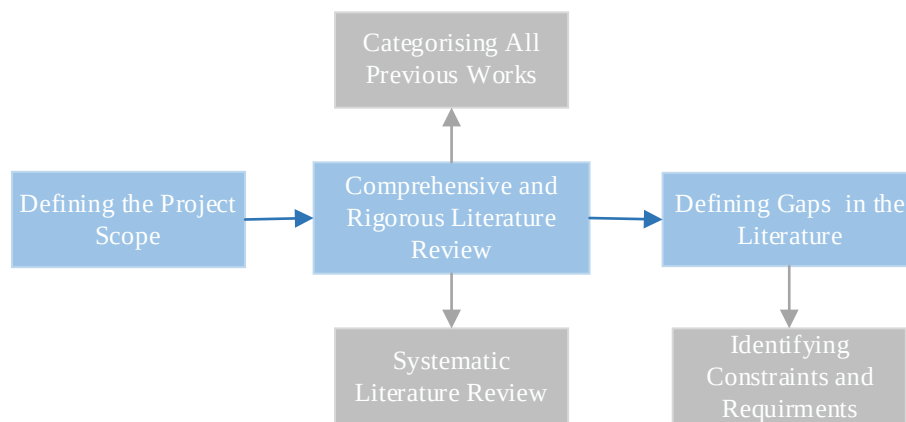


Figure 8 Main steps followed in the project development and the systematic literature review.

This doctoral thesis primarily focuses on the conceptual or early preliminary structural design and mass estimation (SDaME) of the aircraft concepts under study. Hence, the literature about these disciplines has been reviewed in greater depth. In evaluating the different disciplines of strut-braced wing (SBW) aircraft design, the combability of the mathematical model (MM) with other models has been evaluated, and this consideration helped shape the requirements of the model.

2.1 Aircraft Mass Estimation Methods

Aircraft ME is a crucial discipline in the design process of an air vehicle from the initial to the detail design stages, as it directly affects the performance, fuel consumption, and direct operating cost of the air vehicle (Howe, 2000; Dababneh and Kipouros, 2018; Raymer, 2018). The mass of the aircraft is sensitive to several aircraft design decisions. An accurate analysis of the mass growth or decline with the design decisions is essential to achieve the top-level design requirements of an aircraft.

A detailed aircraft group mass statement is given in Table 1 and discussed in the following section. The details of the TOGW breakdown can be seen in Table 1. This thesis will focus on the ME of the wing group and its mass breakdown, as shown in the table below.

2.1.1 Aircraft Group Mass Statement

ME engineers can easily notice that the different institutions use different weight grouping either in their technical reports or in academic studies. Unless the reader pays attention, misinterpretation of the mass reports is possible. Therefore, a clear grouping of the weights of aircraft components is important to avoid any uncertainties before introducing a final ME approach. Hence, in this section, the author presents a clear grouping for aircraft component weights to avoid any uncertainties in developing the new models or when other researchers follow the thesis.

The basic “group weight statement” format is taken from (Raymer, 2018). It is modified to cover all necessary components with a required detail level for this study. In the modification of the format, several other studies are considered to be compatible with their ME methods (Burt, 1962; Torenbeek, 1971; York and Labell, 1980; Macci, 1988; Howe, 1996, 2000; Chiozzotto, 2017; Raymer, 2018; Roskam, 2018).

Table 1 shows the format used in this study. The green boxes are the sum of the mass of subcomponents. When the levels shown in the table's first row increase, the ME approach's detail also increases. Each ME method from the literature is compiled using this table to either compare to each other or combine them with the new methods introduced in this thesis.

Table 1 Weight Groups.

No	Level 1		Level 2		Level3	Level4	Level5	Level 6	Level 7	Level 8	Value
1	1	TOGW	0.00								
2	1	TOGW	1	OEW	0.00						
3	1	TOGW	1	OEW	1Empty Weight Allowance	0.00					
4	1	TOGW	1	OEW	2Structures	0.00					
5	1	TOGW	1	OEW	2Structures	1 Wing Group	0.00				
6	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	0.00			
7	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	0.00		
8	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	1 WB Shear Material	0.00	
9	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	1 WB Shear Material	1 Skin Panels	
10	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	1 WB Shear Material	2 Spar Webs	
11	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	2 WB Bending Material	0.00	
12	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	2 WB Bending Material	1 Spar Caps	
13	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	2 WB Bending Material	2 Stringers	
14	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	0.00	
15	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	1 Store	
16	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	2 Main LG	
17	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	3 Wing Fuel	
18	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	4 Engine	
19	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	5 Wing Fold	
20	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	1 Structure Mass	3 Penalties or Relief	6 Other Con. or Dis. Loads	
21	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Box	1 Structure Mass	4 Rib Mass	0.00	
22	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	2 WB Other Mass	0.00		
23	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	2 WB Other Mass	1 Aeroelastic Penalty	0.00	
24	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	2 WB Other Mass	2 Taper, Joint, Cut-outs	0.00	
25	1	TOGW	1	OEW	2Structures	1 Wing	1 Wing Box	2 WB Other Mass	3 Component Mounting, Bulkhead	0.00	
26	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	0.00			
27	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	1 Leading Edge	0.00		
28	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	2 Trailing Edge	0.00		
29	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	3 Other Sec. Structure	0.00		
30	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	4 LG Doors & Mech.	0.00		
31	1	TOGW	1	OEW	2Structures	1 Wing	2 Wing Other	5 Ail, Elevon and Decel	0.00		

32	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	6	Slats	0.00	
33	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	7	LE Flap Mass	0.00	
34	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	8	TE Flap Mass	0.00	
35	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	9	Spoilers	0.00	
36	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	10	Wing Speed Breaks	0.00	
37	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	11	Winglets	0.00	
38	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	12	Wing Folding Mass	0.00	
39	1	TOGW	1	OEW	2Structures	1	Wing	2 Wing Other	13	Pivot Mass	0.00	
40	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut		0.00		
41	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	1	Strut Box	0.00	
42	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	1	Strut Box	1	Strut Box 0.00
43	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	1	Strut Box	2	Corrections 0.00
44	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	2	Strut Other Mass	0.00	
45	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	2	Strut Other	1	LE Mass 0.00
46	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	2	Strut Other	2	TE Mass 0.00
47	1	TOGW	1	OEW	2Structures	1	Wing	3 Strut	2	Strut Other	3	Strut Mounting 0.00
48	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1		0.00		
49	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	1	Jury1 Box	0.00	
50	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	1	Jury1 Box	1	Jury Box Mass 0.00
51	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	1	Jury1 Box	2	Corrections 0.00
52	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	2	Jury1 Other Mass	0.00	
53	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	2	Jury1 Other	1	LE Mass 0.00
54	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	2	Jury1 Other	2	TE Mass 0.00
55	1	TOGW	1	OEW	2Structures	1	Wing	4 Jury-1	2	Jury1 Other	3	Jury Mounting 0.00
56	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2		0.00		
57	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	1	Jury2 Box	0.00	
58	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	1	Jury2 Box	1	Jury Box Mass 0.00
59	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	1	Jury2 Box	2	Corrections 0.00
60	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	2	Jury2 Other	0.00	
61	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	2	Jury2 Other	1	LE Mass 0.00
62	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	2	Jury2 Other	2	TE Mass 0.00
63	1	TOGW	1	OEW	2Structures	1	Wing	5 Jury-2	2	Jury2 Other	3	Jury Mounting 0.00
64	1	TOGW	1	OEW	2Structures	1	Wing	6 Offset		0.00		
65	1	TOGW	1	OEW	2Structures	1	Wing	6 Offset	1	Offset Box	0.00	
66	1	TOGW	1	OEW	2Structures	1	Wing	6 Offset	1	Offset Box	1	Offset Box Mass 0.00
67	1	TOGW	1	OEW	2Structures	1	Wing	7 Offset	1	Offset Box	2	Corrections 0.00
68	1	TOGW	1	OEW	2Structures	1	Wing	8 Offset	2	Offset Other	0.00	
69	1	TOGW	1	OEW	2Structures	1	Wing	9 Offset	2	Offset Other	1	LE Mass 0.00
70	1	TOGW	1	OEW	2Structures	1	Wing	10 Offset	2	Offset Other	2	TE Mass 0.00
71	1	TOGW	1	OEW	2Structures	1	Wing	11 Offset	2	Offset Other	3	Offset Mounting 0.00
72	1	TOGW	1	OEW	2Structures	2	Horizontal Tail			0.00		
73	1	TOGW	1	OEW	2Structures	3	Vertical tail			0.00		

74	1	TOGW	1	OEW	2Structures	4 Ventral Tail	0.00
75	1	TOGW	1	OEW	2Structures	5 Fuselage	0.00
76	1	TOGW	1	OEW	2Structures	6 Main LG	0.00
77	1	TOGW	1	OEW	2Structures	7 Nose LG	0.00
78	1	TOGW	1	OEW	2Structures	8 Other LG	0.00
79	1	TOGW	1	OEW	2Structures	9 Engine Mount	0.00
80	1	TOGW	1	OEW	2Structures	10 Firewall	0.00
81	1	TOGW	1	OEW	2Structures	11 Engine Section	0.00
82	1	TOGW	1	OEW	2Structures	12 Air Induction	0.00
83	1	TOGW	1	OEW	3Equipment	0.00	
84	1	TOGW	1	OEW	3Equipment	1 Flight Controls	0.00
85	1	TOGW	1	OEW	3Equipment	2 APU	0.00
86	1	TOGW	1	OEW	3Equipment	3 Instruments	0.00
87	1	TOGW	1	OEW	3Equipment	4 Hydraulics	0.00
88	1	TOGW	1	OEW	3Equipment	5 Pneumatics	0.00
89	1	TOGW	1	OEW	3Equipment	6 Electrical	0.00
90	1	TOGW	1	OEW	3Equipment	7 Avionics	0.00
91	1	TOGW	1	OEW	3Equipment	8 Armament	0.00
92	1	TOGW	1	OEW	3Equipment	9 Furnishing	0.00
93	1	TOGW	1	OEW	3Equipment	10 Air Conditioning	0.00
94	1	TOGW	1	OEW	3Equipment	11 Anti-Icing	0.00
95	1	TOGW	1	OEW	3Equipment	12 Photographic	0.00
96	1	TOGW	1	OEW	3Equipment	13 Load & Handling	0.00
97	1	TOGW	1	OEW	3Equipment	14 Mis. Equipment & We	0.00
98	1	TOGW	1	OEW	4Propulsion	0.00	
99	1	TOGW	1	OEW	4Propulsion	1 Engine(s) Instal.	0.00
100	1	TOGW	1	OEW	4Propulsion	2 Accessory Drive	0.00
101	1	TOGW	1	OEW	4Propulsion	3 Exhaust System	0.00
102	1	TOGW	1	OEW	4Propulsion	4 Engine Cooling	0.00
103	1	TOGW	1	OEW	4Propulsion	5 Oil Cooling	0.00
104	1	TOGW	1	OEW	4Propulsion	6 Engine Controls	0.00
105	1	TOGW	1	OEW	4Propulsion	7 Starter	0.00
106	1	TOGW	1	OEW	4Propulsion	8 Fuels System/ Tanks	0.00
107	1	TOGW	2	Useful Load	0.00		
108	1	TOGW	2	Useful Load	1Crew	0.00	
109	1	TOGW	2	Useful Load	2Fuel-usable	0.00	
110	1	TOGW	2	Useful Load	3Fuel-trapped	0.00	
111	1	TOGW	2	Useful Load	4Oil	0.00	
112	1	TOGW	2	Useful Load	5Passengers	0.00	
113	1	TOGW	2	Useful Load	6Cargo/Payload	0.00	
114	1	TOGW	2	Useful Load	7Guns	0.00	
115	1	TOGW	2	Useful Load	8Ammunition	0.00	
116	1	TOGW	2	Useful Load	9Misc Useful Load	0.00	
117	1	TOGW	2	Useful Load	10 Notlisted-1	0.00	

2.1.2 Classification of the Mass Estimation Methods

Before commencing the review of the wing ME works for SBW and TBW aircraft in the literature, the classification of the wing ME methods might prevent possible confusion.

The general terminology for classifying the models was previously discussed in Section - 1.10 (What is a Model?). The structural sizing and mass estimation (SSaME) method aimed to develop in this thesis was classified as the intermediate fidelity level physics-based MM. A more detailed classification can be defined following the terminologies used in the literature on wing ME methods.

Previously Dababneh and Kipouros (2018) reviewed the ME method classifications of different authors and classified the wing ME methods into three main groups empirical-based (low fidelity), analytical-based (intermediate fidelity), and finite element analysis (FEA) based (high fidelity) methods. The details of other classifications can be found in the Ref. (Dababneh and Kipouros, 2018). The following sections will group the SBW and TBW ME studies under these three categories and review their capabilities.

Moreover, the ME method developed in this thesis is the quasi-analytical (intermediate fidelity level physics-based) based ME method following the terminology in Ref. (Dababneh and Kipouros, 2018).

2.1.3 Strut-braced Wings Mass Estimation Studies

Previously a detailed review of the wing ME methods was presented by Dababneh and Kipouros (2018). In this study, the focus will be given the review of the ME methods developed for or used in the design of the strut- and truss-braced wings and other existing methods will also be discussed when required.

2.1.3.1 Empirical and Semi-empirical Methods

Regression analysis is commonly used to develop empirical and semi-empirical ME methods for conceptualising conventional aircraft using historical data. This traditional approach: however, is not applicable for novel configurations as there is neither enough data nor any data of existing samples. Locatelli et al. (Locatelli *et al.*, 2016) previously stated that the regression analysis cannot be applied to a novel type of airliner such as SBW aircraft. Nevertheless, it is essential to know this type of method since some empirical ME approaches might be utilised for some components of SBW aircraft, such as the secondary components of the wing and mass of other aircraft components, as explained in Chapter - 6 (MASS ESTIMATION and VALIDATION).

One of the first SBW ME is presented by Smith *et al.* (1981) utilising unpublished empirical ME method developed for and validated with cantilever wings. The equations are updated and modified to increase the accuracy and cover the effect of strut presence in wing mass. The SBW ME method presented in this work requires further validation for SBW.

Using a physic-based wing ME model from a previous study in Ref. (Chiozzotto, 2016), Chiozzotto (2017) established the design of the experiment (DoG) and presented a new semi-empirical ME method with the help of least square regression for novel wings including aeroelastic effects. All semi-empirical equations for the ME of the aft-swept and forward-swept cantilever wing and SBW can be found in this reference. After introducing semi-empirical wing ME methods, the author combined them with handbook methods to conduct an MDO study for minimum fuel consumption objective.

The ME method was validated for conventional aircraft with actual aircraft data. Nevertheless, the validations of unconventional SBW and forward-swept (FS) SBW were more challenging; hence the detailed design works in the literature were used for validation. While a significant error was obtained in the ME of strut and jury with an error of +110.2%, the rest of the components' ME was close to the reference data.

The results showed that, surprisingly, the new FS SBW configuration showed the best performance for constrained and unconstrained wing-span design studies over the SBW and cantilever wing.

The new method is based on fixed materials for aluminium and composite, and the use of different materials is not possible with the new method. There are other assumptions made to simplify the method of Chiozzotto (2016), and those can be found in the Ref. (Chiozzotto, 2017) to be aware of the limits of the semi-empirical method. The author recommended the implementation of flutter in future works.

2.1.3.2 Analytical and Quasi-analytical Methods

Analytical and quasi-analytical methods are mainly based on sizing the structural components to resist the critical stresses without permanent deformation or failure under different loading cases specified by the airworthiness requirements. The components' critical buckling stress, shear, bending and torsional resistance are primarily used for sizing constraints. After sizing the structural elements, their mass can be computed with material density. These methods suit conceptual and preliminary design studies, and they might be developed to study novel aircraft configurations with novel technologies.

Aircraft structural sizing methods are well explained in several engineering textbooks (Bruhn, 1973; M. C.-Y. Niu, 1988; Niu, 1997; Howe, 2004; Megson, 2013). Nevertheless, those textbooks' existing structural sizing and stress analysis methods lack integration and are presented as disjointed techniques rather than cohesive applications or packages. Therefore, there is a need to meticulously assemble and integrate these methods into an automated sizing model, as demonstrated in this doctoral dissertation.

In 1979, Jobe, Kulfan and Vachal presented the use of a quasi-analytical method in one of the first MDO studies on an SBW and its comparison to a cantilever wing aircraft. The design study was run for a 10,000 *nm* range of freighter with 350,000 *lb* payload and take off field length limit of 90000*ft*.

In this study, flutter constraints were not evaluated. The aircraft was designed with 350 cure T300 graphite/epoxy composite material. ORACLE software was used for the structural analysis and analytical mass calculations. 2.5g manoeuvre and 1.67g taxi load conditions were considered. Statistical and analytical ME methods and their effects on the calculated values of life cycle cost, fuel consumption and TOGW were compared.

It was shown that the empirical ME methods underestimate the structural benefits of SBW configuration. Hence this prevents observing the advantages of SBW over cantilever wing in optimisation studies with objectives of minimum TOGW and fuel consumption.

The SBW configuration prevented the wing-tip deflection causing the touch of the wing to the ground during taxiing. Other advantages of SBW were observed using the analytical ME method. The SBW configuration reduced fuel consumption, gross weight, and empty weight by 1.6%, 1.8%, and 3%, respectively.

The author suggested more detailed structural and aerodynamic analyses to investigate the uncertainties in the presented method.

Naghshineh-Pour (1998) studied the MA of SBW aircraft by modelling the wing box with a double-plate analogy as shown in Figure 9. Three loading cases were applied to the structure, including the symmetric manoeuvres and taxi. With the model, wing bending material weight is computed theoretically. Strut buckling was the critical case for strut sizing under a -1g load case; hence a telescoping sleeve mechanism was introduced to avoid strut buckling. Compared to the cantilever wing competitor, %9.2 TOGW and %18.2 wing weight savings were stated in addition to the %15.4 saving in fuel weight

F. Gern *et al.* (2000) and Naghshineh-Pour (1998) presented one of the first structural sizing and MA of SBW with a telescoping sleeve mechanism used to activate the strut only in tension loads under a 2.5g loading case. Three loading cases of 2.5g manoeuvre, -1.0g pushover and -2.0g taxi bump were covered. In addition to the double plate method used in wing structural ME shown in Figure 9, F. Gern *et al.* (2000) also presented an idealised hexagonal wing box model to compute the torsional stiffness with better accuracy.

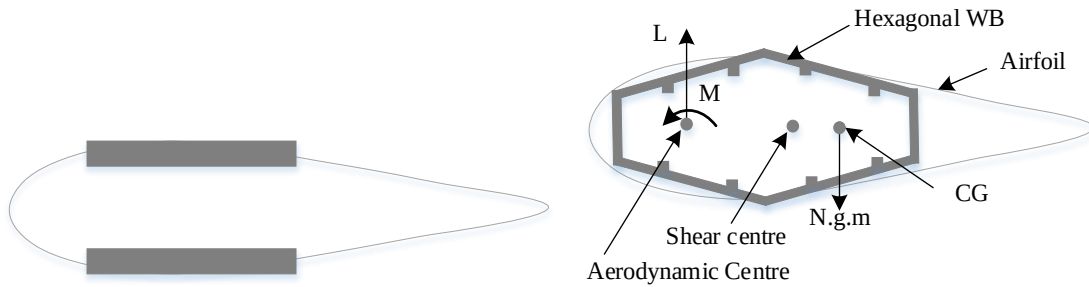


Figure 9 Double plates (left side) and hexagonal wing box (right side) models used by F. Gern *et al.* (2000) and Naghshineh-Pour (1998) for structural analysis and wing mass estimation.

F. Gern *et al.* (2000) aimed for a flexible wing analysis; hence the hexagonal model is stated to cover the wing twist and torsional stress more accurately. The aerodynamic module employed the VLM approach, and its mathematical derivations are explained. The new MA method in Ref. (F. Gern *et al.*, 2000) is validated with data from an existing aircraft and MA estimation results of other methods.

Significant reduction in bending-resistant material mass and 40% wing mass reduction are presented with a strut compared to the cantilever competitor. Additional mass reduction is also shown with the chordwise position of the strut-wing joint. The importance of flexible wing modelling is emphasized to explore the full potential of SBW configuration.

In addition to the SBW ME studies, the double plate model has also been extensively used in ME of the cantilever wings in the literature (Tatham, 1953; Torenbeek, 1992; Elham, 2012; Elham and Tooren, 2013; Elham, La Rocca and Van Tooren, 2013). To increase the method's accuracy, the outer shape of the airfoil is related to the double plate model with the use of the effective distance parameter. The effect of this parameter on the model accuracy is discussed by (Elham, La Rocca and Vos, 2011; Elham and Van Tooren, 2014). Hence, Elham, La Rocca, and Van Tooren (2013) presented an

updated version of the effective distance parameter and achieved better accuracy in wing mass estimation compared to the previous approaches.

Locatelli *et al.* (2016) presented an ME method for SBW aircraft using quasi- analytical based (intermediate fidelity level physic-based) approach following the method in Ref. (Mallik, Kapania and Schetz, 2015), which overestimates the bending stress as it uses fixed panel size. The new method utilises a rectangular box model with mass booms, as seen in Figure 10. Thus, a more accurate ME was achieved. The method mainly relies on sizing the structural components to resist the bending moment, shear force and torsion without a failure for 18 flight cases covering the symmetric manoeuvring, gust, +2g taxi and landing. Apart from those, the deflection constraint was also considered in the wing sizing procedure.

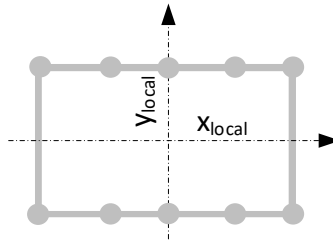


Figure 10 Idealised beam wing section used in the MA method presented by Locatelli *et al.* 2016.

The developed method was integrated into the Pacelab APD framework to compare the MEs of one empirical and one physic-based (using double plates) method using Boeing 737 and 777 cantilever wing aircraft. The method was stated to provide reasonable accuracy in MA. There was no comparative study of the new method with SBW aircraft data. Further validation of the method for SBW is required. Moreover, this method does not entirely estimate the SBW or TBW by covering all the structural components. Some constraints like buckling and aeroelasticity are also not considered in the work.

At the 5th CEAS Air&Space Conference, Chiozzotto (2015) presented an ME method for SBW aircraft considering static aeroelastic effects. Following the same approach with some improvements, the author published a paper in 2016 (Chiozzotto, 2016). Due to the similarities of both papers, except for some minor differences, the last version of the work is reviewed in this section.

The method considers the buckling criteria, fatigue criteria and static aeroelastic constraints such as aeroelastic divergence and aileron reversal, and it was stated to be simple enough to be coded in and used with spreadsheet software. The aeroelastic model was derived following a handbook method for cantilever wings. The modification of the aeroelastic method for SBW was achieved by the author and presented. The method was based on the discretised wing geometry in sections and bays. Ten sections were stated to be enough for adequate accuracy. The discretised geometry form was chosen as it is compatible with aerodynamic VLM, providing better accuracy than the strip theory (Chiozzotto, 2016).

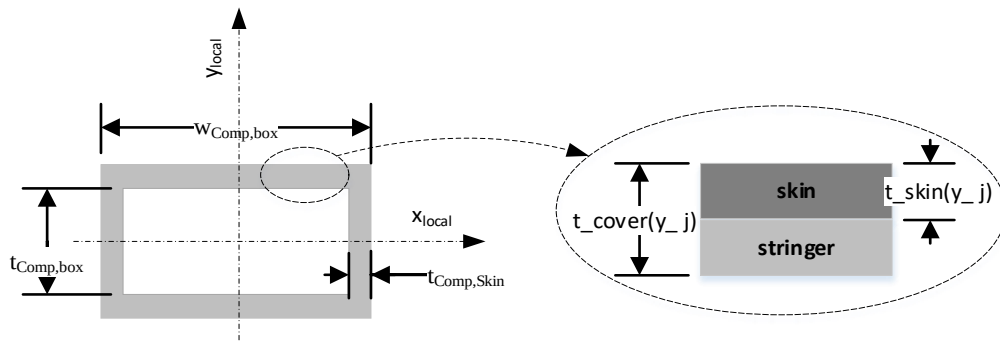


Figure 11 The rectangular wing box model used in (Chiozzotto, 2016).

Chiozzotto (2016) chose the simplified direct approach among the iterative and direct calculation approaches to computing the reaction force in the indeterminate SBW structure problem following Ref. (Pertuze *et al.*, 2010). The strut reaction force was calculated with a simplified approach as explained in (Park, 1980). Structural sizing procedures were conducted using the wing box model given in Figure 9 for the ME of the wing. Nine loading cases were considered in this study involving symmetric and asymmetric manoeuvres, gust, taxi bumps and fatigue.

Regression analysis was used over five different aircraft mass data for the ME of the secondary structures. A method presented by Gur *et al.* (2011) was modified and used for the empirical ME of the folding mechanism. A semi-empirical method was developed to correct the ME of the covers, webs, and ribs by including the joints, standard gages, holes, fasteners, panels and others using the breakdown data of ten aircraft.

The results of the methods were compared to NASTRAN for the validation of the aeroelastic and strut reaction force calculations, to actual aircraft data for the validation of the ME of cantilever wings and the results of a detailed design study (Bradley and

Droney, 2012) for the validation of the SBW ME methods. Twenty-two aircraft were used to validate the ME, and an average absolute error of 10.6% and a maximum error of 22.7% were observed.

Including aeroelastic constraints in the ME method, Chiozzotto (2016) showed the importance of the strut-wing attachment point in terms of aeroelastic efficiency and wing mass. For instance, it was shown that the strut attached to the front spar and at 50% spanwise direction resulted in very similar aerodynamic loading to that of the cantilever wing. A cantilever aircraft with an AR of 20 was shown to be 60% heavier than the baseline model, while the SBW model with an AR of 20 was just 20% heavier than the baseline. However, it should be noted that this study only considers the single disciplinary ME analysis.

In summary, the presented ME method was stated to be accurate enough for the conceptual design studies of the aircraft with SBW by the author. The importance of aeroelastic analysis at the conceptual design level was highlighted.

In another study, Wells, Horvath and McCullers (2017) presented some ME equations. The process employed in the well-known FLOPS computer program for the initial aircraft design at NASA Langley Research Center. A statistically modified physics-based SBW MA equation is presented in this work together with other MA methods for the masses of other components.

2.1.3.3 Finite Element Analysis-Based Methods

Finite element analysis-based ME methods rely on optimising the structural components to resist the stresses computed using finite element analysis to estimate the structural mass. These methods can be utilised with high accuracy in the ME of the novel configurations. However, they usually suffer from high computational time, which might cause significant issues in design exploration studies. The difficulty in their implementation and use in MDO environments was discussed in Section - 1.12 (The Challenge in the Model Development). It will also be evaluated in this section by reviewing the previous studies.

Felix Dorbath and Björn Nagel (2012) presented the details of a wing MA toolchain which estimates the mass of the primary wing structures and the secondary structures, including movables and flap tracks, using FEA. The work introduced the sub-modules used in the tool for structural analysis, aerodynamic analyses, model generation and their interactions.

Bhatia *et al.* (2012) improved a previously developed MDO tool by including flutter constraint and buckling analysis. The derivations of the MMs are explained in the paper for unsteady aerodynamic analysis and structural analysis tools. A rectangular model similar to the illustration in Figure 10 was used in the structural analysis and ME. There were convergence and high computational time problems faced in the study due to the complexity caused by the optimization studies. Hence a linear polynomial response surface is used to simplify the model for a limited design space.

It was shown that the new buckling analysis approach, instead of the simple beam Euler buckling approach provided a lighter structure; however, the lighter structure was prone to flutter, and the inclusion of flutter analysis at the initial design space is suggested. Future works are stated to focus more on convergence and computational time problems.

By coupling the aerodynamic and structural analysis tools, Zhang *et al.* (2012) investigated the effects of the truss on the structural behaviour and mass of SBW configuration. Topology optimization was conducted running FEA in the commercial software ANSYS. An I-beam model was chosen for the structural idealisation as depicted in Figure 12; hence the torsional effects were not covered in the analysis.

SBW with telescopic sleeve mechanism, one-jury, two-jury or three-juries were compared. The SBW provided the best mass reduction with three juries and a sleeve mechanism. The SBW, with the three juries, also performed the best in resisting the fatigue caused by the flutter. This configuration provided an 18% increase in the span length without a mass penalty compared to the cantilever wing baseline.



Figure 12 I beam model for bending mass calculation.

This study did not evaluate the I-beam model accuracy in representing the structural behaviour of SBW. Although the introduction of some valuable tools, there were some uncertainties in the methodology when criticising the potential of SBW. Moreover, the SBW configuration with three juries was found as the most efficient configuration in this study; however, this outcome contradicts other studies in the literature (Bradley and Droney, 2011, 2012; Droney, Harrison and Gatlin, 2018).

Dorbath (2014) presented a physics-based wing ME method. The wing's primary and heavy secondary structures are calculated using a high-fidelity physics-based approach. A finite element shell model achieves stress analysis, and the aerodynamic loads are calculated with VLM. The method is validated against two actual transport aircraft data. The geometric model of the tool enables the design of the novel wings. Five novel aircraft design studies are presented in the work. It was shown that the computation time was 24 hours when full MA capabilities of the tool were used for one loading case with an Intel Core i7-2600, 3.40GHz, 8GB RAM computer. This time can be considered relatively high for conceptual design studies.

2.1.3.4 Suitable Mass Estimation Methods for SBW

Torenbeek (1971) presented a semi-empirical ME method sensitive enough for parametric design studies. The method achieved a standard deviation of 9.64% based on wing mass data of 46 aircraft. The method is just derived for aluminium material. Although SBW configuration was covered in this study, it includes a simple correction factor which only relies on the non-dimensional span-wise position of the strut-wing attachment point.

Dababneh (2016) proposed a wing ME method using MDO. The method is based on FEA with commercial NASTRAN software and a gradient-based optimisation method in NASTRAN. The method was developed for novel aircraft configurations such as; box-wing and blended-wing aircraft. Although this study did not consider the SBW configuration, its similarity to the box wing might enable the MA method for SBW with minimal modifications.

Potter (2017) presented a ME method and software (ASPER) developed for SBW aircraft in Bombardier. The method relies on physics-based formulations. The details and derivation of the methods were not presented in the report. ASPER was validated for wing box ME of SBW configuration using FEA based method; however, there was a discrepancy when ASPER was used for conventional aircraft. ASPER was also implemented in conceptual MDO tools, and the results were compared with other empirical MDO methods. The empirical MDO approach showed some uncertainties. ASPER presented higher values for fuel burn and operating costs than the empirical methods. The report showed the need for further improvements of ASPER, and the author also recommended the aero-structural modelling of the wings as a future work.

2.1.3.5 Concluding Remarks and Discussion

The previous studies on the ME of SBW and TBW aircraft are reviewed in the previous section. It is noted that most of the methods lack accuracy and suffer from long computational times. Hence, the development and implication of accurate enough rapid ME methods are aimed in this thesis.

There has not been a complete ME method covering all components of the TBW configuration. Chiozzotto (2017) presented a complete method; however, the method's average error value was relatively high. Moreover, the error value in the ME of the strut and jury was also high. Considering those gaps, accurate enough and complete ME models are presented in this thesis for jury and strut.

There have not been any detailed methods presented in the literature for the SDaME of the offset in an SBW and TBW configuration. Therefore, methods are developed for offset in this thesis.

No study on the SBW with distributed, electric, or hydrogen-powered propulsion systems has been observed. A more detailed wing box model might be required to cover these technologies, which will be developed in this thesis.

All the load-carrying box models are also reviewed. A more detailed model is developed to study the effect of the wing box design parameters on the wing structural mass, and its use will be explained in the following section.

It can also be observed that each study considered a reduced number of loading cases, and there has not been a precise study showing the dominant loading cases which drive the structural sizing of the wing components. Therefore, the uncertainty will be clarified in Chapter - 8 (CRITICAL LOADING CASES INVESTIGATION).

2.2 Structure

In this section, the studies on the structural analysis of the SBW are reviewed.

A method for simultaneous analysis of wing structure and aerodynamics is presented by Drela (1989). The method is capable of predicting: structural and aerodynamic loads in different flight cases, aerodynamic induced drag analysis with wing twist effect, wing lateral stability derivatives considering the forces that emerged by coupled roll and yaw manoeuvres, divergence and reversal speeds and buckling loads for braced wings. An example of a wing braced with a wire is analysed and tested with the new method.

The challenges and constraints in the structural design of SBW and TBW aircraft were presented by Bhatia *et al.* (2009) using two structural analysis methods of beam idealization using a double plate model (as the sample is shown in Figure 9) and finite element-based structural analysis approach which includes aeroelastic constraints using NASTRAN and its optimisation modules. For the case study, +2.5g and -1g loading factors with a safety factor of 1.5 were used. The calculations were made for Mach 0.85 and 500.000 ft.

Parametric studies were presented to evaluate the effect of wing half-span, wing mid-chord sweep angle, and strut mid-chord sweep angle on the structural behaviour of SBW. Their effect on the aeroelastic characteristic of the wing was observed. An increase in the wingspan, for example, decreased the flutter speed. The calculated wing mass value differed considerably depending on the choice of the rigid or flexible wing box model.

The difficulties in the use of high-fidelity FEA and CFD analysis due to the automatic mesh generation and execution of the analysis in the conceptual design of novel aircraft configurations are studied by Hwang, Kenway and Martins (2014), and the authors present the solutions to these problems. The Boeing SUGAR TBW configuration was analysed to show the capabilities of the new methods.

Kim *et al.* (2015) researched the in-flight application of fibre Bragg grating (FBG) sensors placed on the main spar of a wing at the manufacturing stage for strain measurements and the health monitoring system of a lightweight SBW aircraft. Although there was no transparent qualitative verification of the outcomes, comparing the results gathered from different flight cases, a comparative verification was presented. The presented method gathered data for a total of 74 flight cases.

The elementary structural model of the SBW wing is studied by Dym and Williams (2015) to quickly investigate whether SBW has structural advantages over the cantilever wing before starting the complex analysis. The advantages of SBW configuration were observed and suggested to be studied with more sophisticated design methods.

Cavallaro, Demasi and Passariello (2012, 2014) studied the fully nonlinear stability analysis of joint wings, as the linear methods are unreliable for accurate buckling analysis of SBW. The study investigated the mechanism causing the instability in the structure. The significant effect of bending-torsion coupling due to the wing swept, anisotropic material and the lower-to-upper wing stiffness ratio on the snap buckling are shown. The rotation of the offset (or joint) about the wing-span axis is essential to provide insight into

the stability behaviour. The snap-buckling was also very sensitive to the bending moment transferred from the offset to the upper wing.

Szczyglowski *et al.* (2016) presented a non-linear stress analysis and structural sizing method using a rectangular model similar to the model in Figure 11. The method is verified against cantilever wing data. The effect of piecewise shape functions for the angular deformation of wing sections was studied in detail and presented. An SBW wing from the SUGAR project was tested with the model and compared with the baseline data. It was stated that the non-linear model does not significantly affect the wing's stiffness. There are discrepancies in the ME of the new method compared to data of SUGAR aircraft. The model was suggested to be improved with future works.

Ruffini *et al.* (2020) used Real-Time Hybrid Simulation (RTHS) for the first time to test the gust load mitigation devices of a flexible SBW in a low-speed wind tunnel test. The mitigation device was simulated through a real-time feed-forward feedback controller, and its reaction was applied to the strut via an electro-mechanical linear actuator. Discrepancies between the actual and simulated displacement at the joint of the wing and the subsystem were tried to be prevented using a narrow-band feed-forward displacement control scheme.

Detailed design of strut-fuselage, wing-fuselage carbon-epoxy composite joints of lightweight, two-seat SBW aircraft was studied by Vargas-Rojas (2019). The work reviews different methods for composite material design and manufacturing processes. It presents the synthesis of these methods for more efficient composite material design following the numerical (FE) analysis with Abaqus and ANSYS.

2.2.1 Redundant SBW Problem

To solve the redundant SBW structure problem, Park (1980) derived a simplified beam model of SBW using the Euler-Bernoulli Beam Equation with an assumption of zero deflection at the wing-strut joint point. The assumption was later verified, showing its minor effect on the calculated vertical strut reaction force. An iterative approach was used for the second moment of inertia to find the final thickness of the wing material.

Locatelli *et al.* (2016) solved the redundant structure problem of the SBW configuration with an iterative approach providing deflection compatibility at the strut-wing joint to obtain the strut reaction force. In this study, the strut-wing fixture was chosen as hinged;

the effect of this design decision will also be discussed in Chapter - 7 (PARAMETRIC and SENSITIVITY STUDIES).

2.3 Aerodynamic Research

This section will review the research on the aerodynamic analysis of the SBW.

2.3.1 General Studies

Sellers, Singer and Leavitt (2003) reviewed and discussed aviation industries' challenges and their possible solutions. As a part of this work, transonic and supersonic SBW configurations were covered. Two transonic SBW concepts were discussed, including the fuselage and wing-tip mounted engine configurations. The SBW configuration with the wing-tip mounted engine had a larger vertical tail surface area caused by one engine-off case. This issue was addressed with additional circulation control jets in the vertical tails. The advantages of SBW configuration in reducing the noise level, TOGW, fuel consumption and increasing the lift-to-drag ratio were observed. A possible 30% to 40% increase in the lift-to-drag ratio is stated when an arrow shape SBW configuration is used in supersonic flight. The requirement for further MDO works is presented.

Gur, Schetz and Mason (2010, 2011) researched the effect of aerodynamic design parameters and technologies on the TBW aircraft design using an MDO tool. The drag breakdown is presented in this study as illustrated in Figure 13.

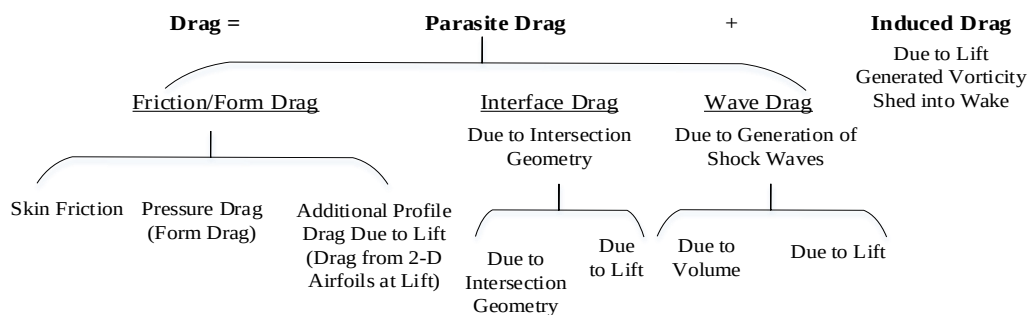


Figure 13 Drag breakdown (Gur, Schetz and Mason, 2010, 2011).

The study's outcome was not quantitative and is briefly summarized in Table 2 and Table 3. The study presented the advantages of TBW configuration over the conventional wing. Due to increased AR and decreased chord length, the two-dimensional cruise lift coefficient appears as a constraint; hence new airfoil designs are recommended. Moreover, the importance of the supercritical airfoil design for TBW configuration is emphasized.

Table 2 The effect of the aerodynamic design parameters on the fuselage design of an aircraft with TBW.

Changing Lifting Surface Parameters	Increased	Decreased
Increasing Laminar Length Ratio	L/D, Fuel Weight, TOGW	Fuselage Friction Drag
Use of the Riblets	<i>No noticeable effect</i>	
Use of Goldschmied Apparatus	L/D	Vehicle drag, Fuel Weight
Tailless Configuration		Fuel Weight
Use of Goldschmied Apparatus and Tailless Configuration	AR	Fuel Weight, TOGW, Wing Area

Table 3 The effect of the aerodynamic design parameters on the wing design space of the aircraft with TBW.

Changing Lifting Surface Parameters	Increased	Decreased
Increasing Natural-laminar-flow Technology Factor	L/D, AR	Fuel Weight
Decreasing Quarter-Chord Sweep	Wave Drags, Friction Drag	L/D
Increasing the Maximum Chordwise Laminar Extent Limit	L/D, AR	Wing Chord, Fuel Weight
Increasing the Maximum 2D Cruise Lift Coefficient	AR, $c_{L,max}$	Fuel Weight, Wing Chord
Increasing Korn Factor	Span, Fuel Weight	Wing sweep, Wave Drags
Decreasing Cruise Mach number	Wing Area	Wave Drags
Use of Winglets	Friction Drag, Wetted Area	Induced Drag
Increasing Fairing Factor Increases	Interface Drag, Fuel Weight	

2.3.2 Interface Drag Studies

Tetrault, Schetz and Grossman (2000, 2001) investigated the interface drag at the strut-wing joint in a transonic flight regime. The viscous and inviscid Spalart-Allmaras turbulence model was used with unstructured meshes. The effects of the incline between the strut and the wing, Reynolds number, and thickness to chord ratio of the strut were investigated.

The angle between the strut and the wing was found to be an essential parameter. The interface drag grows significantly with the reducing angle values. The results showed that the viscous effect alleviates the interface drag by reducing the shock wave and flow separation. Although the thicker profile caused a slightly smaller interface drag for a 90

incline, the interface drag caused by the thicker strut airfoil rises significantly. It exceeds the values of the thinner profile for smaller incline angles between the strut and the wing.

With data from the CFD studies, the author presented a formula in Eq.(2-1) to calculate the interface drag in design studies using the response surface procedure.

$$\begin{aligned}
 C_{D_{interf}} = & 0.1112 - 0.2572 \sin \theta + 3.440 \left(\frac{t}{c} \right) - 0.02097 \log_{10} Re_c \\
 & + 0.09009 \sin^2 \theta - 2.549 \left(\frac{t}{c} \right) \sin \theta + 0.03010 \log_{10} Re_c \sin \theta \quad (2-1) \\
 & - 0.1462 \left(\frac{t}{c} \right) \log_{10} Re_c
 \end{aligned}$$

where $\frac{t}{c}$, Re_c , θ are thickness to chord ratio, Reynolds number (based on the strut chord) and strut-wall incline angle, respectively (Tetrault, Schetz and Grossman, 2000, 2001).

The interface drags at the wing-strut joint are studied with high-fidelity CFD tools by Xiong, Fugate and Nguyen in 2019; and Xiong, Nguyen and Fugate in 2021. First, FUN3D, a high-fidelity CFD tool, was validated with experimental data obtained using the Boeing SUGAR TBW aircraft (Xiong, Fugate and Nguyen, 2019). After the validation, the interface drag effect is investigated at Mach 0.8. It was observed that the strut joint causes abnormal pressure coefficient distribution on the wing's lower surface due to the emerging aerodynamic sucking. In the last stage of the study, the low-fidelity aerodynamic analysis tool VSPAERO of NASA is corrected for TBW interface drag for future studies.

Numerical aerodynamic analysis work is conducted at Virginia Tech University (Hari, Schetz and Kapania, 2019). The drag caused by the strut-wing interference in the supersonic flight regime was investigated. The effect of the angle between the strut and the adjoint surface on the interface drag was examined. Commercial ANSYS Fluent solver was used to solve Navier-Stoke equations, and Spalart-Allmaras was chosen as a turbulence model. It was shown that when the angle of the strut surface decreased, the generated drag decreased. This study was the first to aid the design of SBW supersonic transports by considering strut-wall interaction. Further investigations on shallower intersection angles and swept struts were recommended.

The interface drag between the strut and the wing is studied by Liu and Yu (2021) using CFD with the Spalart-Allmaras turbulence model. The study named the space between

the wing and strut joints 'armpit' region. The effect of the cross-section of the armpit region on the interface drag is considered essential. It was also stated that there is no correlation between the interface drag and the supersonic flow at the armpit region.

2.3.3 Induced Drag Studies

Gagnon and Zingg (2014) previously introduced an Euler-equation-based aerodynamic optimisation model for inviscid, compressible analysis and optimization of unconventional aircraft configurations. This method is used by Gagnon and Zingg (2016) to compare the induced drag reductions in novel configurations of blended wing body, box-wing, SBW and conventional aircraft using nonlinear aerodynamics. Optimized SBW aircraft performed the best-induced drag reduction of %40.3 compared to the baseline cantilever wing model, followed by %34.1 and %36.2 reductions achieved by the box-wing and C-tip blended wing body competitors. The authors stated the inclusion of viscous effects, structural and other design aspects for more accurate comparison in future works.

2.3.4 Extensive Analysis of Drag

2.3.4.1 High Fidelity Studies

A compromise between interface drag, lift-induced drag and the structure's weight defines how successful the new design is. Secco & Martins (2018) used high fidelity Rans-based CFD method to optimise the shape of SBW configuration. A 15% drag reduction was achieved by eliminating shocks and flow separations at the strut-wing junction and minimising the induced drag with HAR SBW. The aerodynamic shape optimisation for minimising interface drags at the strut-wing junction provides a 6% drag saving compared to a baseline design. Overset meshing and component-based parametrisation methods were presented in this study to deal with the occurrence of the negative volume cells during the mesh deformation for optimization.

2.3.4.2 Low Fidelity Studies

Luciano *et al.* (2018a, 2018b) extended a previous physics-based work for induced drag calculations of multi-wing systems to analyse the TBW occupying closed-wing systems with an offset, corners and intersections.

After deriving the MM to calculate the induced drag of SBW, some results obtained with the model were validated with the open-source Athena Vortex Lattice software. Six main

research questions were investigated in this study (Luciano *et al.*, 2018a, 2018b), and the answers to the research questions are as follows;

- Load distribution in a TBW is not fixed - it can vary between the upper wing, offset, and lower wing. This finding questions any preconceived notion of a 'one-size-fits-all' load distribution pattern for TBWs.
- Even among varying load distributions, the maximum aerodynamic efficiency is consistent, representing a universal optimum. This result emphasizes the flexibility in TBW design, with infinite configurations attaining minimum induced drag.
- Pylons typically carry a load, while juries usually remain unloaded or carry a constant aerodynamic load. Any non-constant load on the juries adversely impacts induced drag.
- All optimal solutions stem from a reference point where juries carry zero loads. Thus, a TBW, from an induced drag viewpoint, mirrors the performance of an SBW. Including the juries, however, allows for more load redistribution possibilities without an induced drag penalty.
- The research confirms a universal principle: increasing the AR decreases induced drag.
- Although dihedral introduction benefits stability, it's not ideal for minimizing induced drag. This result underlines the trade-offs designers must consider in wing system configuration.

2.3.5 Other Low-Fidelity Studies

The Tornado open-source MATLAB VLM method code (Melin, 2000) is a suitable aerodynamic analysis tool for the conceptual design process. The code can be modified for different applications, such as flow analysis in transonic and supersonic flight regimes as the correction and modification of the code will be discussed in the following parts referring to the works conducted by (Cheung *et al.*, 2019; Nguyen *et al.*, 2019; Song *et al.*, 2014).

A variable geometry raked wingtip novel control effector (VGRWT/NCE) was examined by Jingeleski, Schetz and Kapania (2013) using Tornado VLM aerodynamic analysis tool (Melin, 2000). The accuracy of Tornado was stated to be sufficient for this research by comparing it to the inviscid fluid analysis results from ANSYS Fluent. It was found that the upswept (VGRWT/NCE) increases the total lift of TBW during the landing and take-

off, whereas the swept configuration offers better drag reduction in the cruise flight. The importance of the (VGRWT/NCE) and wing joint design was emphasized due to the possible drag penalties. The effect of (VGRWT/NCE) on wing pitching moments was also significant, and its use in aeroelastic alleviation or roll control systems was also suggested.

Aerodynamic performances of non-planar wing configurations, including SBW, are studied by Takahashi and Donovan (2011) using an inviscid aerodynamic model coupled with an optimizer. The effect of the strut height to wingspan ratio on the Oswald efficiency factor was tested with minimal values from 1.01 to 1.04.

Conceptual-level aerodynamic analysis of full TBW aircraft is studied by Ting *et al.* (2014) using VORLAX, which is based on the vortex lattice method. A new superposition approach is tested, which mainly analyses each lifting surface of an aircraft individually and sums their aerodynamic results instead of analysing a complete aircraft. The superposition approach showed a good agreement compared to high-fidelity CFD results. Error in lift curve slope, for example, reduced from 40% to 6% compared to the conventional VLM approach analysing the complete vehicle.

Xiong, Bartels and Nguyen (2021) studied the effect of variable chamber continuous trailing-edge flap on the drag reduction of SBW using a modified VLM of VSPAERO. It was shown that the more sections the flaps have, the more the drag is reduced; however, when the weight of actuators is included in the model, the flaps with fewer sections presented the best drag reduction.

The effect of the winglet on the SBW is studied by Metkowski and Maughmer (2021), and a minor 3% increase in the lift-to-drag ratio was observed in the cruise flight. The lift-generating strut concept provided no additional aerodynamic benefit; however, its positive effect in high Mach numbers was stated as it reduced the shock waves.

Xiong, Bartels and Nguyen (2021) modified their VLM tool of VSPAERO to cover the small transonic disturbance, integral boundary-layer, and wing-strut interference. The tool is validated against wind tunnel data.

2.3.6 Other High-Fidelity Studies

Aerodynamic optimization of The Boeing SUGAR SBW Aircraft is studied by Ivaldi *et al.* (2015) using high-fidelity CFD with RANS. The details of the model are not explicit in the paper. The RANS instead of the Euler-based method was chosen to achieve a more

realistic analysis. The SBW configuration is compared to Boeing 737-900ER. Although some drag reduction was achieved with the optimized SBW configuration, the final lift-to-drag ratio of SBW was not better than that of the cantilever competitor. The authors explained the possible limitations of the work which caused these unexpected results. Coarse mesh quality, absence of aero-structural analysis, and coarse junction geometric designs in optimization were the limitations of the study, and they were noted to be studied in the future.

A script-based CAD modelling, named Pascal, was developed by Rajnarayan *et al.* (2019) for aerodynamic analyses at the initial or preliminary design stages. Using the Python library Pascal, a 3D SBW model is created and presented in the paper.

2.3.7 Concluding Remarks

To conclude, the literature on low-fidelity aerodynamic analysis of TBW aircraft points to a common recognition of the utility of VLM-based tools and the importance of the careful design of the wing and control effectors. Nevertheless, contradictions regarding the most accurate approach for analysing the aircraft and the role of juries in achieving optimal induced drag configurations warrant further investigation. The repeated emphasis on the trade-offs in optimizing for multiple performance characteristics also underscores the complexity of TBW aircraft design and the value of comprehensive, multidisciplinary design approaches.

The literature on the high-fidelity aerodynamic analysis of SBW aircraft suggests a consensus on the significance of strut-wing interface drag and the need for further high-fidelity aerodynamic analysis, specifically incorporating viscous effects, structural considerations, and aero-structural analysis. However, some contradictions emerge regarding the impact of strut design parameters, such as the angle and thickness-to-chord ratio on interference drag, necessitating further research for clear insight. Refining mesh quality, incorporating aero-structural analysis, and developing low-fidelity aerodynamic tools also present promising avenues for future investigation.

2.4 Fluid-Structure Interaction

Xiong, Nguyen and Fugate (2020) and Nguyen, Xiong and Fugate (2021) use an in-house developed tool coupling the VLM tool (VSPAERO) to a finite element tool (BEAM3D) for Fluid-Structure Interaction (FSI) studies. VSPAERO is updated to cover

the transonic and viscous effects and strut-wing interface aerodynamic correction. It is validated against wing tunnel data and high-fidelity CFD models with FUN3D.

Single and multi-point optimization studies were run for three flight cases with three different lift coefficients at 40,000 ft. It was shown that there is a minimal reduction in the drag coefficient achieved through the single-point optimization (Xiong, Nguyen and Fugate, 2020), while the multi-point optimization provided a higher reduction in drag (Nguyen, Xiong and Fugate, 2021).

Vedernikov *et al.* (2021) presented a decomposition approach of structural analysis, strength task and loading cases to reduce the computational time and labour work in the preliminary air vehicle design. It was achieved by an algorithm which adjusts the balance between the accuracy and complexity of the methods by deciding the fidelity level automatically. At least a 20-times reduction in computational time was stated to be achieved. The method was validated against more detailed conventional design methods. The method is named a four-level algorithm (FLA). Its capabilities are also shown in the design of SBW configurations by studying the optimum strut-wing joint location considering the weight as an objective.

2.5 Propulsion, Fuel Consumption, and Cost Estimation

The exigence of reducing fuel consumption and increasing aircraft miles per gallon led to the research question of whether SBW could provide advantages over cantilever wings in 1980. Park (1980) studied the SBW configuration involving the aeroelastic constraints in the model. As a result of the project, it was stated that the SBW configuration does not provide an advantage as it consumes almost the same amount of fuel due to the additional weight and parasite drag caused by the strut.

However, it should be noted that the work was not conducted in an MDO environment. The study without MDO might limit the evaluation of the potential of SBW. In the study, for instance, an AR of 14 was chosen for the SBW configuration; nevertheless, it was shown in recent research that the SBW configuration provides benefits for higher ARs in fuel consumption.

Chakraborty *et al.* (2014) studied the optimisation of fuel consumption and direct operating cost (DOC) of one-jury SBW aircraft for the Mach numbers varying from 0.60 to 0.80. The optimised baseline was chosen from the previous study of Nam *et al.* (2014),

and the MDO environment presented by Grasmeyer (1999) was used for the investigation.

Although it was found that fuel consumption increases with increasing Mach numbers, total DOC tends to decrease with increasing Mach numbers. The significant effect of depreciation explained the cause of this, and crew costs as they increased with longer flight time for the same distance. Surprisingly, there was no convergence between the minimum fuel consumption and the minimum total DOC value. However, it should be noted that the assumptions and empirical approaches (based on conventional aircraft data) might cause uncertainties. A further study with more robust and higher fidelity methods was recommended.

The ATUGA SBW configuration of Cranfield University is evaluated in formation flight to reduce fuel consumption by Durango, Lawson and Shahneh (2016). The safety aspect of flight distance and other aspects such as structural fatigue, sensors and passenger well-being were also covered as constraints in the study. It is noted that SBW configuration, which benefits mostly from laminar flow surfaces, compared to previous studies with conventional cantilever wing aircraft, provided a lesser reduction in fuel burn in formation flight. The author concluded that SBW aircraft are not promising enough for formation flight, considering the additional complications they cause during a mission.

Lee, Nam and Kang (2017) proposed a propulsion system modelling approach for TBW aircraft to explore wider design space. The model was derived with a thermodynamic cycle using numerical propulsion system simulation (NPSS).

The advantages of the proposed method are the inclusion of the engine installation effects and the technology factor counting the effect of future technologies in the engine-airframe design space for conceptual design studies. For the efficient integration of propulsion and airframe design models, the reduced order model (ROM) of the new NPSS was used in design space studies. Moreover, an artificial neural network (ANN) was also used to estimate the effect of engine components on their weight.

The design exploration study was later run with the proposed propulsion-airframe design framework. It was found that the models ignoring the effect of engine installation can overestimate the performance of TBW aircraft. Furthermore, the new approach also showed the increased performance of TBW with reduced block fuel by 14.75%. The reduction even further improved with the inclusion of the effect of engine technology

factor. The author suggested studying novel engine configurations, such as hybrid electric-gas engines, in future work.

2.6 Aeroelasticity of SBW

This section reviews the aeroelastic studies on the SBW and TBW aircraft. The studies are grouped according to their objectives, research questions and methods.

2.6.1 Low Order Physics-based Aeroelastic Models

In 2017, Su developed a model to analyse the aeroelasticity and flight dynamics of SBWs. The study's objective was to propose a method for SBW optimization at the conceptual multidisciplinary design optimization (MDO) stage and design the structure to prevent coupling between flight dynamics and the nonlinear aeroelasticity behaviour of highly flexible aircraft. A low-order strain-based geometrically nonlinear aeroelastic equation was integrated with a rigid-body motion to assess the impact of the strut attachment point on SBW's aeroelastic behaviour.

The results indicated a significant decrease in flutter speed when the strut was added to the wing, attributed to the strut modes, as the strut rigidity was only 10% of the wings. Furthermore, longer strut lengths (i.e., 70% of the half-wingspan) led to lower flutter speeds due to increased flexibility. Thus, a softer strut was suitable for shorter strut lengths, while increased strut stiffness corresponded to increased flutter speed for longer struts.

Another observation was a minimal increase in flutter speed, about 5%, due to SBW and rigid body coupling. Strut positions above 70% of the half-wingspan demonstrated higher flutter speeds, rendering the strut addition effect negligible. However, the derived aerodynamic model for this study did not account for compressible flow or transonic effects (Su, 2017).

An 8.5% drag reduction was reported from 40% to 70% strut positions in aerodynamic performance analysis. However, the aerodynamic model only considered the induced drag. The strut inclusion also reduced significant wing-tip deflection from 28.5% of the wing span to 3.5% (Su, 2017).

In conclusion, Su stated that the developed model was sufficiently precise and versatile for aeroelastic analysis in conceptual MDO studies.

A comprehensive work was conducted by Nguyen et al. (2019) and Fugate et al. (2019) to investigate the flutter behaviour of transonic SBW and to optimise its flight jig twist. The authors used tools developed by the researchers at NASA and presented in two conference papers at AIAA SciTech 2019 Forum.

Both papers describe the developed aero-structural model for TTBW using open-source VSPAERO software. A steady-state aerodynamic analysis model was created using the Vortex Lattice Method (VLM) in VSPAERO. The model was then modified with transonic and viscous flow corrections to capture viscous and transonic flow regimes using a transonic small disturbance tool TSFOIL coupled to another in-house integral boundary layer code. Another correction was implemented for analysing the wing-strut intersection using an in-house high-fidelity CFD code of FUN3D. Roe's flux-difference splitting scheme and Spalart-Allmaras turbulence model were utilised in CFD code, and experimental data validated the results.

Furthermore, an in-house FE code of BEAM3D was used for dynamic structural analysis. The FE model comprises a geometric nonlinearity to account for a deflection-dependent nonlinear stiffness caused by the tension in the strut. The aerodynamic and the FE models were coupled to conduct the aero-structural and flutter analyses of TBW aircraft.

As a result of the research, it was stated that the proposed correction methods provide an accurate, low-cost solution for the steady and unsteady analysis of the TBW aircraft concept compared to high-fidelity, computationally expensive CFD methods. For further research, the flight dynamics of TTBW aircraft with the help of a developed optimisation tool suggested by the authors.

2.6.2 Higher Order Methods

Vos *et al.* (2020) reveal the details of aeroelastic analysis tool development from static aeroelastic analysis to a dynamic analysis using the open-source FEA tool of B2000++ and the in-house Navier Stokes Multi Block (NSMB) CFD tool. The new method is validated with the real aircraft data and used in the AGILE project's aeroelastic analysis of SBW aircraft. Aeroelastic constraints reduced the lift and drag coefficient of SBW aircraft; however, the aerodynamic performances improved for higher Mach numbers.

2.6.3 Parametric Studies

Sulaeman et al. (2002) investigated the aeroelastic behaviour of SBW with strut attachment point variation along the wing. MSC/NASTRAN was used for the analysis.

An idealised hexagonal wing box model of Lockheed Martin Aeronautical Systems (LMAS) was used to create the FEA model. The paper might be considered one of the most detailed ones to evaluate and introduce the hexagonal wing idealisation model of LMAS for wing ME and structural analysis.

Some of the results obtained in this study are comparable with more recent ones. This study reported that the flutter speed is almost levelled off for the strut attached between 10% and 70% of the wing span length and reduced when the joint approaches the wing tip. A similar observation was presented by Su (2017). However, different reasons for this behaviour were described by the authors. While Su (2017) attributed the reduced strut stiffness to increased strut length, Sulaeman et al. (2002) stated that this is due to the dominant effect of bending moment at the inboard region of the wing; hence it causes higher flutter speed in those regions.

Sulaeman et al. (2002) also highlighted that the flutter speed in the tested cases is higher enough than the flutter envelope defined according to regulations. It should be noted that there is no clear and comprehensive verification of the study in this paper.

TBW structures commonly feature a half-span of around 170 feet, resulting in highly flexible wings due to reduced wing-chord and thickness-to-chord ratios. Bhatia, Kapania and Haftka (2012) analysed the effect of key parameters on the aeroelastic behaviour of TBW aircraft, such as the wing's half-span, strut sweep, spanwise intersection point, and the number of truss elements in the setup. These configurations are sized based on strength and assessed for aeroelastic performance.

The findings show all the examined parameters have significant implications. In most scenarios, extending the half-span elevates wing weight while lowering natural frequencies and flutter speed. A larger wing-strut sweep angle disparity increases wing weight but positively affects the flutter speed. The ideal spanwise intersection lies between 55% and 70% for optimal wing weight and flutter speed. These insights are poised to inform future truss-braced-wing design optimization studies.

Meinicke, da Silva and Guedes (2019) conducted a parametric flutter analysis on a medium-range aircraft flying at 0.85 Mach and 35,000 feet, using the One-Factor-At-a-Time (OAT) method with MSC NASTRAN. They examined parameters such as wing AR, engine position along the spanwise direction, length of strut offset, strut and wing sweep angles, wing-strut joint in the spanwise direction, wing taper ratio, and the influence of pre-stressing the strut.

Their findings suggested that flutter speed tends to decrease with increasing values of AR, length of the offset, strut-wing intersection point towards the wing-tip, and wing taper ratio. Notably, in the strut-wing joint point analysis, the strut surface area remained constant, leading to a reduced strut chord and stiffness as the strut span increased.

In contrast with Su's (2017) study, they found different influences of the strut-wing joint on flutter behaviour. Also, flutter speed increased as the engine position moved towards the wing tip (particularly at the strut-wing joint) and as the wing sweep angle increased. However, there was no discernible pattern in flutter speed variation with changes in strut force and sweep angle. The authors caution against using these findings as primary design drivers.

2.6.4 Aeroelastic Alleviation

The SBW and TBW configurations have been extensively investigated in the context of active and passive aeroelastic alleviation methods. This section categorises the active aeroelastic alleviation techniques into variable geometry and load alleviation dampers.

2.6.4.1 Variable Geometries

Chen *et al.* (2015) researched the application of variable camber continuous trailing edge flaps (VCCTEF) for active aeroelastic control (AAC) and drag reduction in TBW aircraft. They chose Boeing's Sugar-High model for their evaluations and optimizations.

Commercial software was utilised for structural, aerodynamic, and aeroelastic analysis. This software included aeroelastic tailoring of composite skins to meet aeroelastic constraints. Based on the findings from the VCCTEF modification and its optimised deflection for drag reduction and load alleviation, the TBW baseline was iteratively optimized.

The results showed that the flutter speed of the baseline model reached the flutter boundary. However, the VCCTEF modification increased the flutter speed to 1.15 times higher than the nearest flutter boundary point without additional weight penalty. Furthermore, they optimized flap deflections for minimum drag during +1g cruise speed, achieving a drag reduction of 11%. This saving equates to an annual fuel saving of 128,000 lbs per aircraft.

It is worth noting that this study was not conducted within a full MDO, encompassing all realistic constraints for TBW. As a result, there may be discrepancies in the findings, and the presented potential might be slightly reduced.

Active flutter and gust alleviation using variable-camber continuous trailing-edge flaps were studied by Bartels, Stanford and Waite (2019) utilising Boeing transonic TBW aircraft designed for Mach 0.745.

A linearized reduced order model was created from an unsteady RANS simulation for aerodynamic analysis. Open and closed-loop control models were developed and tested. It was found that the closed-loop control model was successful in suppressing the aeroelastic issues and stabilising the aircraft against gusts. Evaluation of control surfaces distributed along the wing indicated that the inboard, mid, and outboard wing flaps were necessary along with the horizontal stabiliser to respond to the gust. The reduced order aerodynamic model was also verified with the higher order CFD simulation to predict the flutter behaviour; however, there were some discrepancies in the pitch and plunge behaviours of the models to gust.

2.6.4.2 Load Alleviation Dampers

The aeroelastic behaviour of NASA/Boeing SUGAR Bolt aircraft was analysed using MSC Nastran by Szczyglowski *et al.* (2017) to investigate the load alleviation dampers, which are linear, rotational, viscous, and mounted at the strut-fuselage and strut-wing attachment points. It was shown that the dampers performed better at the strut-fuselage joint compared to the strut-wing joint attachment. Additionally, further improvement was observed using dampers at both joints. In this study "1-cosine" gust case is used, and the critical effect of gust frequency on the worst loading case stress levels was also observed.

Passive gust load suppression systems for SBW were studied by Szczyglowski *et al.* (2017) using commercial FEA software MSC NASTRAN, and later this work was expanded in Ref. (Szczyglowski *et al.*, 2019). Three different positions, including the wing-fuselage, strut-wing and jury-wing joints, were investigated as a location of load alleviation devices. Three devices, only damping and two other inerter-based designs, were also investigated.

It was found that the strut-fuselage joint showed the best performance as a device mounting location. The mounting of the devices to the jury-wing connection showed an insignificant effect, while the strut-wing point showed a minor effect. Moreover, single and multiple-device designs were also investigated. For a single device case, a pure damper device at the strut-fuselage joint achieved the highest spanwise bending moment reduction by 3.8% at the wing root. Instead, the inerter-based device showed a similar

reduction with a lower viscous damping coefficient requirement. In investigating multiple device designs, two inerter-based devices provided a spanwise bending moment reduction of 4.2% at the wing root and identical alleviations for the other five inboard beam loads.

2.6.5 Aeroelastic Tailoring

Wang *et al.* (2015) studied the preliminary design of an SBW aircraft using aircraft 765-095-TS1 N+4 from Bradley and Droney (2012). The project aimed to investigate the fuel-saving potential of the SBW aircraft with the objective of minimising aircraft wing weight while considering the structural and aeroelastic constraints together with aeroelastic tailoring. Those constraints include the Mach range of 0.6~0.8, the strain of +0.004 for aluminium alloy and -0.004~0.006 for composite structure. The loading cases were 2.5g pull-up and -1.0g push-over with the aileron deflection of $\pm 25^\circ$. It was observed that the SBW configuration could achieve a significant maximum fuel saving of 35% with a transition point of laminar flow located at 50% wing chord.

After introducing a reduced order aeroelastic model including gust loads for fatigue analysis in the author's previous work (Rajpal, Gillebaart and De Breuker, 2019), Rajpal and De Breuker (2019) investigated the effect of fatigue in the structural sizing of composite SBW and its weight. This study chose A medium-range SBW aircraft as a baseline with a range of 3500km, a maximum take-off weight of 3900kg, and a 0.78 Cruise Mach. Other information on the configuration can be found in the paper.

Two aeroelastic optimisation studies were performed using PROTEUS (the in-house code of Delft University), which is improved by integrating the reduced order and analytical fatigue models. Five thousand three hundred sixty load cases, including static and dynamic loads, were tested. Two optimisation studies were performed using first a dynamic aeroelastic without a fatigue constraint and second one with fatigue constraint. The results indicate that both the dynamic and the static loading cases participate in the structural sizing of SBW.

It was shown that the optimisation study with a fatigue constraint provided a 22% heavier wing structure compared to the first study without a fatigue constraint. The study generally shows the importance of fatigue analysis in the structural design of composite SBW.

2.6.6 Experimental Works

An experimental wind tunnel test on the aeroservoelastic behaviour of TBW was presented by Scott *et al.* (2016) under the Boeing SUGAR project. Two main aims of the research were identifying the boundary of open-loop flutter and achieving its suppression and indicating gust relief. Open-loop flutter and limit cycle oscillation (LCO) inception limits were obtained for a series of Mach numbers and angles of attacks. The flutter easing control laws were tested for gust load relief using a Transonic Dynamic Tunnel. The evaluation of the aeroservoelastic model generated from acquired data, the experimental equipment and methods with the results can be reached from the paper. Two successful control law models were presented for alleviating the model in flutter speed and LCO. As a result, the importance of including the control laws in the full-scale-TBW design stages was shown in this study.

A scaled at 4.5% TBW model with the average wing sweep and wing span of 12.5° and 7.7 feet, respectively, were tested in the 11-by-11-ft transonic wind tunnel by Drain *et al.* (2017) to investigate the bending and torsion deformation of the wing. Three configurations, including a cantilever wing, SBW without an offset and with an offset, were tested. While the strut without an offset design was attached to the wing LE point, strut attachment with an offset design varied along the wing's local chord. Stereophotogrammetry is used to measure deformations. Issues occurred in the location and design of the measurement targets on the wing, and they are stated to be the main source of measurement discrepancies. It was stated that although wing twist was not affected by the presence of the strut, the SBW design with an offset achieved 30 to 40% more stiffness in the reduction of bending deformations.

The aeroelastic behaviour of the Boeing SUGAR TBW aircraft is tested in the NASA Langley Research Center Transonic Dynamics Tunnel by Bartels, Funk and Scott (2017). The wind tunnel model experienced classical flutter and limit-cycle oscillation (LCO). The LCO and flutter dynamic pressure and Mach number onset increase with increasing angle of attack because of tension/compression of the strut.

2.6.7 Other Studies

The effect of pre-stressed elements in an SBW configuration was investigated by Coggin *et al.* (2014) and Zhao *et al.* (2015). The angle of attack, aerodynamic weighting factors to capture transonic effects in VLM, and ballast mass are estimated to affect pre-stressed

structural elements; hence their effect on a TBW configuration designed for the NASA transonic dynamics tunnel was studied using a NASTRAN non-linear aeroelastic tools.

2.7 Flight Dynamics

Su (2017) developed an MM capable of evaluating the flight dynamic and aeroelastic behaviour of SBW. Although the derived equations were coupled to these two sub-disciplines in the paper, only one case study was presented for the aeroelastic analysis of the SBW configuration. The developed method was explained enough to be implemented in MDO conceptual design tools. Section - 2.6.1 (Low Order Physics-based Aeroelastic Models) provides a more detailed review of the work.

The coupling between the structural degrees of freedom and rigid body due to the highly flexible nature of the wings of novel aircraft and the possibility of significant effect of the static deformations on the inertia tensor were investigated by Gupta *et al.* (2018) in the flight dynamic analysis. Gupta *et al.* (2018) proposed a mathematical formulation to calculate the first and second-order inertia effects and the second-order momentum-coupling vectors (the density-weighted volume integrals) with just the consistent mass matrix of the aircraft FEM model without access to the FEM source code. The formulation was then verified with a simple cantilever wing model under a uniformly distributed load with exact analytical mode shapes and assumed modes.

After verifying the formulation, the inertia coupling terms of SBW and TBW configurations, which were previously designed in the Virginia Tech MDO environment at Mach 0.7 and 0.8, were analysed with a 2.5g loading factor. Only the wing and strut of the aircraft were modelled in this study without a rigid body. A higher flexibility effect was observed in the SBW configuration compared to TBW at Mach 0.7. However, an opposite trend was shown when the configurations were analysed at Mach 0.8. The analyses of the entire aircraft were proposed to be a future study by the authors.

2.8 High-Fidelity Conceptual Design Methods

Böhnke (2015) utilized a surrogate model based on symbolic regression to increase the computational speed and accuracy at the conceptual design stage for novel configurations. Automated multi-fidelity tool being able to switch different fidelity level methods is achieved. SBW configuration was also effectively tested in the new tool.

Anil Variyar and Thomas D. Economon (2016) studied the conceptual design of the SBW aircraft using physics-based methods, which were mostly high-fidelity. As a conceptual

design tool, open source SUAVE (Lukaczyk et al., 2015; Timothy et al., 2017), written in Python and C++, was integrated with high-fidelity CFD tool SU2 (Economon, Mudigere, et al., 2016; Economon, Palacios et al., 2016), FE analysis tool NASTRAN and the automated mesh generation tool GeoMACH (Hwang and Martins, 2012). Other tools used in this study can be found in Table 4.

Table 4 Methods and tools used in work (Anil Variyar and Thomas D. Economon, 2016).

Sub-Discipline	Name of Tool / Method	Programming Language	Fidelity Level, If Possible
Conceptual Design	SUAVE	Python	Low
Aerodynamic	SU2	Python, C++	High
Aerodynamic	Vortex Lattice		Mid
Mass Estimation & Structure	NASTRAN		High
Mass Estimation & Structure	TACS		High
IT	pyOpt, VyPy		
Optimisation	Surrogate (GPR)		

Instead of running an optimisation study for all flight cases, only +2.5g and -1g loading cases with MTOW were considered. Data from NASA CRM (the μ CRM configuration) was also used to validate the study, and the SBW aircraft of NASA SUGAR Project Phase II (Bradley and Droney, 2012) was chosen as a baseline model.

There are no general results or constraints to be used in developing a new conceptual design methodology in this paper. The paper mostly aimed to describe the developed methodology and the tools. The research can benefit future studies by describing all the open-source tools used.

Although the paper describes a high-fidelity design optimisation framework and tools for SBW configuration, it might be controversial to label the method as a suitable tool for conceptual design studies to explore the design space of SBW aircraft. The methodology presented here might require a longer time than conventional simple and quick conceptual design tools, and the method might be evaluated as either a preliminary design optimisation tool or an approach between conceptual and preliminary design stages. There were no aeroelastic constraints in this study. The proposed method was limited to exploring the design space of the SBW configuration as it could not consider

crucial design variables such as; the strut-wing joint location in chord-wise and span-wise directions. Although CFD was used in this study to generate aerodynamic data more accurately, it was not used to evaluate the effect of strut-wing interface drag or optimisation.

2.9 Multidisciplinary Design Optimisation Studies

Several MDO works of SBW and TBW aircraft have been presented in the literature by different organisations. This section reviews the previous works and groups them under relevant sub-titles which indicate the project or the organisations which conducted the works.

2.9.1 ONERA's ALBATROS Project

In 2010, ONERA introduced the ALBATROS (Aile Laminaire haute à Traînée Réduite par Optimisation multidisciplinaire) project, which aims to investigate and present the potential advantages of SBW aircraft configuration to motivate future research. In their article in 2012, Carrier et al. provided an overview of the project describing the design approaches and aerodynamic, structural, aeroelastic and flight mechanic investigations of a chosen SBW baseline aircraft.

2.9.1.1 Design Methods

In that study, a conceptual design work was introduced, and the novel aircraft design methods the formulae taught in the aircraft design module of Engineering School (ISAE-Supaero) in France in the 1990s. The ONERA-led ALBATROSS project (Carrier *et al.*, 2012) embarked on the investigation of a 180-passenger SBW configuration featuring a nominal range of 3000 nm, cruising at Mach 0.75, and powered by twin rear fuselage-mounted turbofans.

Their strategy encompassed three main stages: an initial phase relying on semi-empirical formulas for preliminary design, primarily focusing on fuel burn and DOC optimization related to engine sizing and cruising altitude; a subsequent phase employing sophisticated aero-structural simulations along with a handling quality analysis to gauge the specific advantages of the SBW design; and lastly, a parameter-driven study focusing on strut curvature, strut-wing connectivity in both chord and span directions, as well as strut thickness. However, in the first phase of the ALBATROSS project, the design space exploration remained confined to a limited subset, with the crucial task of integrating

various analytical tools into a coherent MDO framework deferred to a subsequent, dedicated segment of the project.

2.9.1.2 Structural Analysis

Another MDO analysis of SBW aircraft structures considered various methods for selecting the laminate stacking angles, rib spacing, stiffener spacing, and strut design. According to Gundlach IV *et al.* (2000), selecting the laminate stacking angles was a critical aspect of the structural design, with considerations also given to the additional weight incurred by incorporating copper sheets into the composite layers.

An interval of 0.4m to 0.7m was recommended for rib spacing following guidelines provided by Tetrault, Schetz and Grossman (2001). While multiple approaches exist for stiffener spacing, Carrier *et al.* (2012) opted for the analytical method delineated in Ko, Mason and Grossman (2003).

The strut was modelled as a single section in an FEA under a hard landing scenario with a -2g loading factor. This approach asks whether this method was intended to obfuscate potential strut buckling issues. Subsequent application of a beam model in a 6 DoF FE analysis revealed the optimal strut position to be between 50% and 70% for both effectiveness and weight considerations (Carrier *et al.*, 2012).

The concept of a telescoping sleeve mechanism introduced by Frank H. Gern *et al.* (2001) was considered for preventing strut buckling. This design allowed for the strut to be sized for positive loading cases above +0.8g, a measure that averted a total excess weight that would offset SBW aircraft's advantages, as Kulfan & Vachal (1978) proposed. However, Carrier *et al.* (2012) did not incorporate this mechanism but proposed a curved strut to control buckling.

The limitations of the current studies include the potential for an inaccurate comparison of SBW and cantilever wings due to the fixed AR assumption. It is also suggested that empirical formulae used in these studies may not completely apply to the SBW aircraft concept, as there are currently no validated formulae for SBW. Further research is necessary to comprehensively understand these phenomena and enhance the strut design in future studies.

2.9.2 DLR's FrEACs Project

In DLR's internally funded project "Future enhanced aircraft configurations (FrEACs)", Moerland *et al.* (2017) investigated the SBW configuration in a collaborative environment

consisting of several specialists from each sub-discipline of aircraft design. Due to the insufficient existing data or examples of reference SBW configuration, the specialist's implicit design knowledge is shown to be important. Hence, among six different SBW configurations, a rear fuselage-mounted counter-rotating open rotor (CROR) aircraft configuration was chosen as a baseline, and the effect of the SBW was investigated. The top-level aircraft requirements of this study are given in Table 5.

Table 5 TLAR of reference aircraft and SBW win configuration used in Moerland et al. (2017).

Parameter	Reference Aircraft	SBW
Number of passengers	150	154
Maximum payload, <i>t</i>	18.5	18.63
Maximum range, <i>nm</i>	1700	1700
Mach number in cruise	0.78	0.72
Maximum operating altitude, <i>ft</i>	39800	37000
Approach speed, <i>kts</i>	141	141
Maximum wing span, <i>m</i>	36	36
Take-off field length, <i>m</i>	2090	2100

2.9.2.1 Methods and Their Fidelity Level Classification

The methods & software used in the study are briefly listed in Table 6, and the details of the fidelity levels of the methods are also explained in Table 7. The MDO tools were classified into four fidelity levels starting from *Level 0* to *Level 3* as given in Table 6.

Level 0 design tools were used to generate the baseline model with the formula generated from empirical data and surrogate model. VAMPzero, an open source *Level 0* aircraft conceptual design tool written in Python, is embedded with the design methods introduced by DLR previously (Böhneke, Nagel and Gollnick, 2011; Moerland, Becker and Nagel, 2015) and used in this study.

Table 6 Method and Software used in (Moerland et al., 2017).

Tool's Discipline	Tool's Name -Method	Fidelity Level If Available	Direct Link, If Available
Conceptual Design	VAMPzero	Level0	(German Aerospace Center (DLR), no date)
Aerodynamic	Lifting Line	Level1 & Level2	(Liersch and Wunderlich, 2008)
Aerodynamic	HandbookAero		(Raymer, 2006)
Aerodynamic	Korn/ Mason Approach	Level1 & Level2	(Gur, Mason and Schetz, 2010)
Structure	DELIS - FE Analysis		
MA	Physic based approaches		(Chiozzotto, 2016, 2017)
Engine Design	GTlab-Performance		
IT	CPACS		(German Aerospace Center (DLR), 2020a)
IT	RCE		(German Aerospace Center (DLR), 2020b)
IT	IDEaliSM		
IT	AGILE		(AGILE, 2020)

Table 7 Categorising the fidelity of the methodologies in aircraft design and their usage (Moerland, Becker and Nagel, 2015; Moerland et al., 2017).

Fidelity Level	Definition	Runtime per DOE point	Usage in the simulation workflow
L0	Empirical and surrogate models	<1 minute	A foundational approach generates initial geometries and mass breakdowns that align with TLARs and make necessary assumptions.
L1	Analysis using simplified physics-based models	10 minutes	Simulations consist of the essential stage where the entire experimental design is formulated.
L2	Accurate physics-based models	100 minutes	It serves as a foundation to verify and, if necessary, amend the overall findings of L1 analyses, offering a deeper understanding of the underlying physical factors influencing the configuration.
L3	The most accurate physics-based analysis	order of days	Excluded from the analyses as it is deemed integral to the detailed design phases aimed at capturing localized effects.

The *Level1*, physic-based design tools were the core of exploring design space. The Common Parametric Aircraft Configuration Scheme (CPACS) developed by DLR was used to share the output and input data in a common format for all the tools (Liersch and Hepperle, 2011; Nagel et al., 2012; German Aerospace Center (DLR), 2020a). A digital environment to create a simulation workflow was also developed by DLR previously and

used in this MDO work (German Aerospace Center (DLR), 2020b). The previous works published by Chiozzotto (2016, 2017) were used for the ME analysis. There is a need for future design tools that can explore bigger design possibilities by considering more factors, such as shape, size, and functionality, to account for the impact of both geometric and operational limitations.

2.9.2.2 Aerodynamic Performance

Some key points can be extracted from this study (Moerland et al., 2017). It was shown that the advantages of SBW configuration over cantilever competitors could be obtained in longer wing spans over 36m. Hence, it was decided to design the aircraft for larger segment airports with a wingspan of 52m. The most promising SBW contribution was shown with a 12% reduction in fuel consumption compared to its cantilever counterpart. The strut position of 52% wingspan was a compromise between the lower drag of shorter strut length and the weight saving due to bending load relief. Adding one jury was shown to have advantages in terms of weight saving. The ME of the jury was calculated with some statistical and simplified relations (Chiozzotto, 2016).

2.9.2.3 Structural Design

Structural analysis showed that the strut reduced the bending moment significantly; the strut also caused an aeroelastic relief. The importance of the aeroelastic constraints was highlighted, and the divergence speed, aileron reversal and flutter speed were evaluated. The constraints of flutter speed caused an increase of 42% in wing+strut weight. Nevertheless, although this configuration was exposed to be 24% lighter than the cantilever counterpart, divergence did not occur in any configuration. It was also shown that if the structure was designed considering the aileron reversal, the SBW wing configuration might not be beneficial anymore. Hence instead of sizing the wing for aileron reversal, it was decided to find other solutions, such as using the inboard ailerons or spoilers as high-speed ailerons.

Utilising high-fidelity FE analysis with DLR's tool DELIS, it was shown that strut-buckling by axial and bending load is the essential constraint in the sizing of the strut with pinned boundary conditions. Previously, Frank H. Gern et al. (2001) introduced a mechanism which only activated the strut in positive loading cases; this resulted in no strut buckling. The study of Moerland et al. (2017) showed that no strut-buckling case resulted in a 57% lighter strut structure. Hence a clear understanding of the advantages and disadvantages

of fixed or pinned strut-wing joints or even the case of no strut buckling is a requirement and should be investigated.

The effect of torsional stiffness was not a constraint for the wing box design in the SBW configuration due to the low sweep angle of the new configuration. An increase of 1.2% was reported in the mass of the fuselage due to the higher cruise altitude chosen for better aerodynamic efficiency. The handling quality of the SBW configuration was also shown to be adequate in the case of lateral and longitudinal motion.

2.9.2.4 Advanced Propulsion

It should be added that the counter Rotating Open Rotor (CROR) pusher engine was chosen, positioned at the rear fuselage, based on expert preference. The potential effects of the engine's location on SBW design, including aspects like total wing weight and inertia relief, weren't thoroughly examined. However, the paper mentions a reduction in airfoil thickness due to this choice. While the study advocates for using a CROR engine due to its potential benefits, the noise issue of this type of engine remains unsolved and could limit its application in airliners. Therefore, the feasibility of a wing-mounted engine configuration is still relevant and deserves further exploration. Additionally, the MDO investigation focused solely on the selected SBW configuration, potentially limiting the discovery of the complete capabilities of SBW aircraft by neglecting the exploration of alternative configurations.

2.9.3 The Horizon H2020 Funded AGILE Project

The AGILE research project aims to create and formulate a large-scale optimisation and design environment for the collaboration of heterogeneous teams of experts with the latest IT technologies, and it was funded by the Horizon H2020 program (AGILE, 2016). A design methodology, the AGILE paradigm, was previously introduced by Nagel and Ciampa (2018) to provide a design framework for MDO of future aircraft concepts. AGILE received the admired "ICAS Award for Innovation in Aeronautics" with an innovative contribution to aeronautical engineering for their MDO environment, accelerating the speed of realistic MDO projects by 40% (AGILE, 2020).

2.9.3.1 Design Methods

An MDO study of SBW, including aeroelastic concerns, was carried out by the AGILE partners to reduce the fuel consumption of SBW wings to meet the targets mentioned in the 2050 flightpath report of the European Commission (European Commission, 2011).

AGILE paradigm (Ciampa and Nagel, 2017) was used as a framework in this study for the MDO of composite SBW aircraft. The detail of the developed framework for SBW is introduced in another study presented by Torrigiani et al. (2018). The optimisation tools consist of sub-tools from different collaboratives, including VAMPzero - design synthesis; PROTEUS – Aeroelastic code consists of finite element (FE) beam model using geometrically non-linear Timoshenko beam elements coupled to VLM (Werter and De Breuker, 2016); AMLoad – Aeroelastic tool predict the flutter behaviour of the wing and is suitable for conceptual design; ASTRID -subsystem architecture; Engine Deck – selecting engine; Mission Analysis – required fuel; FSI – aeroelastic deformation.

The design framework combines low-fidelity (LF) and high-fidelity (HF) optimization loops. HF aeroelastic tool was initialised for static aeroelastic evaluations using the LF loop. There are iterations between HF and LF loops. The objective of the study was to optimise the wing mass.

2.9.3.2 Surrogate Models

As stated above in the description of the sub-tools, although the proposed design synthesis tool was designed to combine HF and LF analyses, it can be noted that the main structural analysis method is based on the computationally expensive FEA. Hence, a surrogate model of PROTEUS and PROTEUS-AMLoad was designed and used for SDaME (Nagel and Ciampa, 2018); apart from that, a surrogate model of all the workflow was also created and used. The accuracy of the surrogate model was acceptable for wing ME with an error of 2%; however, strut mass and flutter speed estimation had 8% and 14% errors, respectively and need to be improved in future works.

2.9.3.3 Aerodynamic Performance

Torrighiani et al. (2018) presented the results obtained in the AGILE project. Different SBW configurations were investigated in a multilevel and multidisciplinary design environment. This collaborative study includes partners from different countries and organisations. The main design aspects of aircraft design were studied by each group, including onboard systems and cost evaluation. Aeroelastic analysis and engine-wing integration were covered in this study. Full-size aircraft was modelled using Euler simulation in an open-source CFD code (SU2) by TSAGI. The cell numbers of the wing-fuselage and wing-fuselage-nacelle configurations were 3.3 and 5.8 million, respectively. The nacelle involvement was found to reduce the aerodynamic performances by 10%. $0.4 C_l$ and 0° the angle of attack was found acceptable for the configuration. It was also

shown that the strut causes drag due to shock waves as much as that caused by the wing.

2.9.3.4 Results of Improved MDO Framework

In the next step, the framework described by Rajpal *et al.* (2018) is used for MDO studies (Torrighiani *et al.*, 2018). While the PROTEUS code was utilised for aeroelastic tailoring of the composite structure and the ME of the wing box in this study and the companion paper, this study also employed the DLR's tool wiSe for the ME of the secondary airframe components. Apart from that, cost and emission analysis was conducted by RWTH Aachen University (Franz *et al.*, 2012; Franz *et al.*, 2013; Lammering *et al.*, 2012). As the PROTEUS and AMLoad codes are computationally expensive and take one day to analyse ten configurations, the surrogate model of those codes and the surrogate model of all workflow were created (Rajpal *et al.*, 2018).

According to the top-level aircraft requirement (TLAR) and previous studies of Prakasha *et al.* (2016), All-electric Aircraft (AEA) were studied in Ref. (Rajpal *et al.*, 2018). The Ice Protection System (IPS) system was chosen thermal rather than an aerothermal method. Designing an onboard system (OBS) provided reduced mass and reduced required power (Cronin, 1983).

It was also stated that the mass of the block fuel increases with the increasing wing span; however, it decreases with the increasing AR. Therefore considering the required wing area, the optimum wing size can be chosen. Other results considering other disciplines can be found in the attributed article (Torrighiani *et al.*, 2018).

In future works, another HF approach was mentioned, and the strut mass and displacement analysis was underlined to be investigated more. Flight mechanics of SBW aircraft were also emphasised as the HARW is expected to cause handling problems (Torrighiani *et al.*, 2018).

2.9.4 Boeing/NASA SUGAR Project

The SUGAR project of Boeing started as a response to NASA's solicitation to achieve advanced aircraft configurations with new technologies to achieve the targets, including reductions in fuel consumption, noise level and emissions. The projected operation timeframe is between 2030 and 2035. The fourth phase of the project was started in December 2016. The progress in phase 4 and previous ones was summarised (Droney, Harrison and Gatlin, 2018) along with the introductions of future objectives. The detailed

summary of all the phases of the SUGAR project can be accessed in Ref. (Droney, Harrison and Gatlin, 2018); hence they are not repeated in this section; however, when required, relevant publications will be referred to in this thesis.

2.9.4.1 Advanced Propulsion

The Subsonic Ultra Green Aircraft Research N+3 initiative (SUGAR N+3) (Bradley and Droney, 2011) explored various aircraft configurations, including conventional, blended wing body, and SBW structures, projected for the 2030-2035 timeframe. The emphasis was on a medium-sized aircraft model capable of carrying 154 passengers, and comparative analyses were performed for a specified mission range of 900 nm. Findings from the study underscored that integrating the SBW configuration with hybrid propulsion systems emerged as the most fuel-efficient solution. The initial phase of the SUGAR project primarily targeted the implications of incorporating new propulsion technologies. Despite the detailed examination of structural, aerodynamic, and propulsion systems within the design process, only two cycles of the overall multidisciplinary analysis were conducted. This constrained number of iterations could potentially downplay the advantages of innovative configurations. Consequently, advancements in streamlining and automating the design process are warranted to circumvent this predicament.

2.9.4.2 Fuel Consumption

In The Boeing Company funded project as a part of SUGAR, 2.9.5 Virginia Polytechnic Institute and State University (VT) and Georgia Institute of Technology (Georgia Tech) investigated the fuel consumption of SBW configurations without a jury (C100 Series), with one jury (C200 Series) and two juries (C300 Series) considering the different span limits in an MDO environment using ModelCentre's optimisation tools of Design Explorer and Darwin optimisers (Chakraborty *et al.*, 2015).

This work emphasises the trade-off of using pin or ball joints for strut and truss assembly. It can be said that there is still a contradiction in the literature regarding the type of strut wing joints. It was shown that the effect of joint types on structural weight was minor; hence ball joint was chosen for the design considering the easiness in manufacturing, assembly and maintenance. The uncertainty is caused by the fact that the paper does not provide the modified structural MM or its reference used to calculate the effect of different joint members.

VT MDO tool was modified and integrated with new tools during the MDO study. Boeing Company provided the surrogate tail sizing method; however, it was not presented in the

report, and there is no access to the document (Chakraborty *et al.*, 2015). Other modifications to the MDO environment, which were also not presented in this study clearly, were wingtip clearance angle examination, engine position analysis as a design parameter and wing positioning to achieve 25% MAC to an aimed longitudinal position. Apart from those, a modification was made to the ME tool to cover the change of CG position due to fuel burn.

It was shown that adding one jury reduces the wing structural weight, TOGW and fuel consumption considerably. After adding a second jury, minor performance enhancement was achieved but made the system more complex. Therefore, one jury design was chosen. The fuel efficiency was shown to increase with increasing span length as can be seen in; however, after the length of 170ft, there is only a minor reduction in fuel consumption, hence 170ft span length was selected.

2.9.4.3 Aerodynamic Performance

In the second stage of the MDO work, the effect of natural laminar flow on fuel consumption and lift-to-drag ratio (L/D) was investigated by comparing the wing designs with the turbulent flow on both surfaces, NLF on the upper surface - turbulent on the lower surface and NLF on both surfaces. It was shown that for the highest L/D ratio and the minimum block fuel weight, the wing design with NLF on the lower and upper surface provided the best performance. After some modifications, NLF on the upper surface was the best choice for aerodynamic performance and low-speed characteristics. It was stated that the choice of laminar flow option limits the application of LE edge devices (Chakraborty *et al.*, 2015); hence other solutions can be studied for this issue.

The quantitative analysis of the uncertainties using empirical tools and assumptions was recommended as a future research topic.

2.9.5 Virginia Polytechnic Institute and State University

2.9.5.1 Initial MDO Works

Naghshineh-Pour (1998) developed an SBW conceptual design tool capable of accurately predicting this concept's wing mass. Flight Optimization System (FLOPS) and Aircraft Synthesis ACSYNT (Vanderplaats, 1976) were employed for MDO studies.

The bending material mass was calculated using an idealised double plates box (Figure 9), and the method was coupled to Flight Optimisation System (FLOPS). The wing was analysed with +2.5g, -1.0g and a taxi bump of +2.0g loading cases. The study showed

that the -1.0g loading case was critical for SBW structural design, and a telescoping sleeve mechanism was created to activate the strut in the positive loading case while deactivating it in the negative loading case. The SBW structure was analysed as a determined structure in this study. As the author applied different strut reaction loads as input to optimise SBW's structure, the model does not use indeterminate structure analysis.

The SBW concept achieved 9.2%, 18.2% and 15.4% savings in take-off gross mass, wing group mass and consumed fuel mass, respectively, compared with the cantilever counterpart.

The research of Grasmeyer (1999), as identified by Wang *et al.* (2015), represents one of the earliest attempts at a Multidisciplinary Design Optimization (MDO) application to an SBW. Grasmeyer (1999) used the Boeing 777-200IGW as a baseline, optimizing the SBW configurations for minimum Take-off Gross Weight (TOGW) using 18 design parameters and seven constraints. The configurations considered included wing-tip-mounted and wing-mounted-engine designs.

Grasmeyer first assessed the aerodynamic efficiency of a wing-tip-mounted engine based on previous research (Miranda and Brennan, 1986; Patterson and Bartlett, 1987). The study noted a large rudder size for the one-engine-out case and thus proposed the application of an innovative circulation control (CC) system to reduce the rudder size. This approach is particularly pertinent given the recent advancements in CC technologies.

The MDO environment in Grasmeyer's study leveraged simple methods for aerodynamic performance and drag evaluation, including wave drag, interface drag, parasite drag, and Non-Laminar Flow (NLF) analysis. However, the empirical method for interface drag derived from low-speed data necessitates improvement for transonic conditions, possibly through Computational Fluid Dynamics (CFD) or experimental data.

After validating the MDO tool with Boeing 747-100 and 777-200IGW data (Kulfan and Vachal, 1978), the SBW wing-tip-mounted engine and SBW wing-mounted engine configurations, both with an AR of 20.5, showed a reduction in fuel weight by 21% and 29% respectively, and up to 14% and 15% decrease in TOGW compared to the optimised cantilever design. Nevertheless, the study highlighted limitations in the SBW wing-tip-mounted engine configuration, including increased wing-tip deflection during

taxi bumps and larger rudder size. Further, the maximum span length of the HAR SBW configuration is constrained due to airport gate box limitations.

2.9.5.2 Load Alleviation Studies

F. H. Gern *et al.* (2000) used the same MDO framework as Grasmeyer (1999), improving the flexible structural analysis approach and using actual design loads. Similarly, wing, wing-tip and fuselage-mounted SBW configurations were compared to the optimised cantilever for minimum TOGW objective.

The model in this paper uses the telescopic sleeve mechanism to deactivate the strut in negative loadings, such as taxi-bump load at -1.67g. Hence the wing behaves as a cantilever beam in this case. This approach should be evaluated carefully, as it has benefits and drawbacks.

The results show the advantages of SBW over cantilever configuration with reduced fuel consumption. According to the upgraded MDO studies, the cantilever wing leads to a 40% mass penalty to achieve the same thin airfoil of SBW. Moreover, while the MDO computation time of the rigid model took 15-20 minutes, the flexible model corresponds to 2-3 hours. Hence a multi-fidelity approach was taken by running a rigid model and later optimising the configuration with a flexible model to reduce the computation time.

2.9.5.3 Structural Performance

Expanding on Naghshineh-Pour's (1998) research, Frank H. Gern *et al.* (2001) proposed an innovative approach for sizing the structural elements of the wing by considering wing flexibility. They developed an aero-structural model that coupled structural analysis with a linearised transonic vortex lattice method.

Initially, a double plate model was employed to calculate the bending material weight of the wing box. However, when it proved inadequate for capturing torsional constraints, a hexagonal model was also used, and the outcomes were compared. Notably, the study implemented jig-twist calculations in their analysis procedure, which were previously shown to reduce design loads by 10% in HARWs potentially (Gimmestad, 1979).

The study considered two loading scenarios, a +2.5-g manoeuvre and a -1.67 taxi bump. Notably, negative loading during taxi-bump led to strut buckling, necessitating increased strut thickness, consequently diminishing the advantages of the SBW concept. Therefore, similar to the previous, the telescoping sleeve mechanism was employed for

the strut-wing connection, which operates only during positive loading cases. A vertical strut offset was also implemented to lessen the interface drag at the wing-strut junction.

The model's results were validated with Boeing 747-100 data and compared with the Torenbeek method (Torenbeek, 1992). Both flexible and rigid wing box models delivered reasonably accurate results for conceptual design analysis. The study also discovered that wing torsional stiffness was high, and even engine torsional moment had a minimal impact. Comparing a cantilever and SBW design for a HAR thin airfoil wing, the SBW configuration provided 49% weight savings with the rigid-wing model and a 20% reduction with the flexible model. Chord-wise strut position analysis further revealed that strut positioning close to the Leading Edge (LE) resulted in the least structural weight. At the same time, wing box structures increased towards the Trailing Edge (TE) attachment points.

2.9.5.4 1-Jury and 2-Jury Truss-braced Wings

Gur et al. (2011) introduced a new MDO framework by improving the previous MDO environment of Virginia Tech and introducing new approaches. The new environment uses finite elements with idealised rectangular wing box geometry for structural analysis, semi-empirical methods for aerodynamics, and a mixture of empirical FLOPS and physic-based approaches for MA. The FLOPS is utilised for the ME of secondary structures, while the wing box material weight is estimated with a physics-based approach. One simple and one more detailed analysis module in FLOPS was presented for the performance analysis. A hybrid optimisation approach was also presented. This tool first creates a surrogate model to find the global minimums and later occupies a gradient-based approach to find local minimums. Another important improvement, according to the author, was made in the geometric parametrisation plan, which enabled modification of the environment for more complex configurations easily and reduced the computational time by a significant amount.

After introducing the new MDO framework, an MDO analysis was conducted to compare the cantilever wing, SBW, 1-jury TBW and 2-jury TBW for minimum fuel consumption objective. A mission profile of an airliner similar to Boeing 77-200ER with a 7,730 nm range and 305 passengers was chosen. Loading cases of +2.5g pull-up manoeuvre, -1.0 g manoeuvre and 2g taxi bump with gust load were evaluated for structural loading.

2-jury SBW provided the most promising performances with a 20% reduction in fuel consumption compared to the cantilever configuration.

2.9.5.5 Performance Analysis of Transonic TBW Airliner

Gundlach *et al.* (2000) employed a Multidisciplinary Design Optimization (MDO) approach to analyse an SBW configuration in a transonic flight regime in their comprehensive conceptual design study. The researchers selected a mission profile featuring 325 passengers covering a 7500 *nm* range at Mach 0.85 for a 2010 operation entry date. The study optimized SBW and cantilever wing configurations by employing 15 to 22 design variables and 17 constraints for minimum Take-off Gross Weight (TOGW) and fuel burn, using 2010 technology factors.

The results underscored the dominance of the SBW configuration over the cantilever wing, both in terms of TOGW reduction (8.1%) and fuel burn decrease (13.6% - 15%). The study also highlighted the noise reduction potential of the SBW due to a 20.7% smaller engine size. Furthermore, the SBW offered more benefits for longer-range missions, with a maximum range of 8.4% longer than the cantilever wing configuration.

Gundlach *et al.* (2000) indicated the SBW configuration was highly responsive to technological improvements and particularly underscored the benefits of tip-mounted SBW in reducing induced drag. However, they also acknowledged the challenges posed by a one-engine-out constraint. Recommended areas for future exploration included mitigating the drawbacks of the tip-mounted SBW configuration, improving the structural analysis tool, and examining other SBW configurations to understand their full potential.

2.9.5.6 Subsonic Truss-braced Wing Aircraft

With twelve constraints, Gupta *et al.* (2014) studied MDO to compare SBW, TBW (one jury) and cantilever wing aircraft using Lockheed Martin C-130J as a baseline in a subsonic flight regime. Virginia Tech's in-house MDO code was used for this study, and Raymer's handbook method was used for flutter constraints. The configurations were tested under 17 loading cases.

The obvious advantages of SBW over its cantilever counterpart were shown in this study, with 10.7% fuel reduction causing just a 5% increase in TOGW. It was shown that the SBW configuration provided even better performance than TBW, which achieved a fuel reduction of 4.5% with an 11% increase in TOGW. The study's results contradict the previous research (Bradley and Droney, 2012). The author explained the reason for this unexpected result. The flight mission and cruise Mach number might limit the potential of TBW as both configuration results in the same AR. At the same time, adding a jury increases the weight and causes additional interface drag.

It was noted that SBW configuration is a better concept for two inboard engine configurations. The layout of TBW has small space for two inboard engines, affecting the strut-wing joint and causing longer strut length. Adding a second engine also provides reasonable inertia relief for SBW, which was already obtained with one jury in TBW. This statement can be researched in future studies.

2.9.5.7 Transonic Truss-braced Wing Airliner with Flutter Constraint

Building upon previous research of Gern *et al.* (1999), H. *et al.* (2001) presented a comprehensive study on SBW aircraft, targeting minimum Take-off Gross Weight (TOGW) and fuel consumption while considering aeroelastic constraints. The methodologies used involved semi-empirical and physics-based approaches for aerodynamic analysis and offline NASTRAN analysis for flutter constraints.

Regarding structural design and ME, the study employed FLOPS's semi-empirical methods and an idealised hexagonal wing box model from LMAS. This hexagonal model aided in calculating the bending material weight and conducting aeroelastic assessments. A telescopic sleeve mechanism was also applied, similar to previous works.

The configurations under examination comprised a fuselage-mounted engine, high-wing SBW and cantilever variants of low and high wings with wing and fuselage-mounted engines. These designs catered to a long-range mission profile with 325 passengers and 7500 nautical miles range (plus 500 nautical miles fuel reserve), cruising at Mach 0.85. Loading scenarios evaluated included 2.5g manoeuvre, -1g push-over, and -2.0g taxi bump cases.

The SBW configuration demonstrated lower fuel consumption—11.85% and 12.2% less than the cantilever counterpart—for minimum TOGW and minimum fuel consumption objectives, respectively. Remarkably, the study also found that the flutter speed of the SBW configuration was naturally higher than the flutter boundary, conflicting with more recent studies (Mallik, Kapania and Schetz, 2013). However, the research did not account for the potential adverse effect of fuselage-mounted engine configuration on longitudinal stability.

Long-range and medium-range SBW civil airliners were previously designed and analysed in an MDO environment. Mallik *et al.* (2013), in their medium-range airliner MDO study, considered the flutter constraint. The optimisation objective was minimum TOGW achievement. The optimised SBW configuration with flutter constraint was

compared to the optimised SBW aircraft without flutter constraint to evaluate the effect of a flutter on SBW MDO. Another investigation was studied to compare the optimised SBW and cantilever configurations when the flutter constraint is active. The flutter boundary was 1.1 times the dive speed instead of 1.5 considering the possible use of active flutter suppression systems.

First, it was stated that the flutter constraint increased the TOGW of SBW aircraft by 1% while increasing the structural weight by 10%. Second, it was found that the optimised cantilever wing is unaffected by flutter constraint, which diminishes the advantages of SBW over the cantilever wing. Therefore, the author recommended that future studies optimise SBW for minimum fuel consumption objectives to investigate the advantages of the configuration. Finally, the importance of flutter constraint in SBW design was highlighted in this study.

In their 2015 study, Mallik et al. incorporated aeroelastic constraints into an MDO framework to compare the potential of medium and long-range SBW configurations to their cantilever counterparts, with minimum TOGW and fuel consumption as optimization objectives. They used a more sophisticated rectangular wing box model with mass booms.

One key finding was the notable impact of flutter constraints on the total weight and fuel consumption of SBW designs. Despite penalties imposed by flutter speed constraints, the study demonstrated that SBW configurations maintained superiority over their cantilever counterparts. For instance, the medium-range SBW configuration offered a 3.5% fuel consumption reduction over the cantilever design when optimized for TOGW and a 6% reduction when optimized for minimum fuel consumption.

The study further validated the accuracy of a linear aeroelastic MM against a nonlinear counterpart created with NASTRAN, affirming its suitability for conceptual MDO studies. The optimal positions for strut-wing and strut-fuselage junctions for fuel efficiency and minimum TOGW were identified as 60% of the half-wingspan and the wing mid-chord, respectively.

Lastly, the study explored active flutter suppression mechanisms, recommending future research into the trade-off between the weight penalty of these mechanisms and the possible reduction in wing structural weight due to relaxed flutter constraints.

2.9.6 Other Multidisciplinary Design Optimisation Works

2.9.6.1 Fuel Consumption

A commercial aircraft conceptual design tool, PACELAB, mainly using empirical methods, was modified by Riggins (2015) with theoretical physics-based methods for the conceptual design of SBW aircraft.

The aerodynamic, ME and engine scaling modules were upgraded. The aerodynamic calculations were mainly based on the strip line approach with the corrections for transonic flight. Previously published CFD analysis of SBW by Tetrault et al. (2012) was used for the interface drag computation. The aerodynamic model was validated with two different aircraft data. An in-house FEA approach was used to calculate the wing box mass; the secondary structural weight was calculated empirically using FLOPS. A total of 18 loading cases incorporating symmetric manoeuvre, gust and taxi bump were considered for structural sizing.

After validating the design tool, a design study compared the cantilever wing aircraft and its counterpart SBW configuration was conducted. It was shown that while the fuel saving achieved by the cantilever configuration was between 10% and 30%, the SBW configuration provided a fuel saving of 37%.

2.9.6.2 Load Alleviation

Cheung *et al.* (2019) studied the folding wing mechanism and stated that the mechanism provides gust load alleviation by folding the wing during flight with an active control system. An 11% reduction in the peak bending moment value at the wing root was achieved using the mechanism in the experimental wind tunnel tests at Bristol University. This reduction is achieved by giving the hinge an angle to reduce the local angle of attack of the wing tip. By achieving wing-root bending moment reduction, lighter wing box structures can be designed, and more fuel-efficient concepts can be achieved. It was also stated that the mechanism increased the load envelope in some flight conditions.

2.10 Patents

They are also patented ideas which present either solutions to the specific problems of SBW configuration or improve their performance. In a patent document (Loma and Beach, 2018), the Boeing Company presented an SBW design to reduce shock waves between the strut-wing interaction point. The proposed solution is a variable wing

thickness along the span. From the wing root to the strut-wing junction, the wing's thickness decreases first and increases approaching the joint to reduce the shock waves.

Wright (2017) patented a solution for the wing folding mechanism of SBW aircraft. The new folding mechanism approach is different from the classic ones, which generally places the actuator inside the wing, hence the moment arm, which folds the outer part of the wing, mainly cannot be longer than the wing thickness. However, the patented idea is to fold the wing using the outer part of the strut. This outer part may be extendable with an actuator system placed inside the inner part of the strut.

2.11 Strut-Braced Wing Project at Cranfield University

At Cranfield University, four MSc students developed an automated aircraft conceptual design tool, including the analyses of ME (Canosa, 2014), aerodynamics (Sun, 2014), propulsion (Ingles, 2014), and the effects of aircraft layout (Reiser, 2014), to design medium to UHARW advanced aircraft involving SBW and cantilever wings. All sub-modules were combined in an Excel spreadsheet (Reiser, 2014). In this section, the critical review of each work is presented.

2.11.1 Mass Estimation Module

Canosa (2014) developed a wing ME tool combining the theoretically based handbook sizing methods (Bisplinghoff, R. L., Ashley, 1962; Bruhn, 1973; M. C. Y. Niu, 1988; Niu, 1997; Baker, A., Dutton, S., Kelly, 2004). The method was combined with the global aircraft design tool developed by Reiser (2014).

Instead of using the idealised wing box models, the author followed a detailed stress analysis and sizing approach following the aircraft structural design books. Moreover, contrary to most of the methods in the literature, Canosa sized the ribs, leading and trailing edge with analytical approaches instead of using empirical methods. The structural components are sized using only one loading case of +2.5g. The validation and use of these methods should be considered carefully. These detailed approaches increased the computational time above 1 hour to complete one loading case analysis.

The method is verified with data from AT-13 aircraft designed at Cranfield University. The developed model estimated the wing group mass of AT-13 aircraft to be 2.4% higher. Another MA validation study was also run with the Boeing 737-800 cantilever wing airliner, and a maximum difference of 6.3% was observed.

Some discrepancies in the developed MA method can be observed from the detailed results. Although the strain energy-based approach for calculating strut reaction force was suitable, the position of the strut wing joint tended to be close to the wing root according to the results. The researcher defined the strut position as a minimum 20% wingspan length constraint. Hence, the best strut position with minimal weight objective was found around 25% span length. This result contradicts the findings in the literature, revealing that the ideal strut position is about the mid-span length for an optimum wing group mass without aeroelastic constraints.

It was also presented that the total mass of the strut was reduced from 2543kg to 778 kg with the addition of one truss. Although it is known that adding a truss reduces the total mass of the strut-truss structure due to buckling load alleviation, 69% mass reduction contradicts the literature. Therefore, the solution to the redundant SBW structure problem and the truss components' ME methods should be revised and verified.

2.11.2 Propulsion Module

A review of the potential future propulsion systems considering the HARW aircraft was presented by Ingles (2014) using documents from engine manufacturers and publications. The evaluation of the ducted fans with geared and counter-rotating configurations was emphasised to increase propulsion efficiency with a promising noise level compared to the counter-rotating open rotor (CROR) engines. The CROR might be the best solution for fuel efficiency if the noise issue can be alleviated. It was also stated that the hydrogen or electric combined fan is likely to decrease fuel consumption further by 2035.

2.11.3 Aerodynamics Module

The aerodynamic analysis module was developed by Sun (2014). The tool was based on empirical methods. Several empirical handbook methods were evaluated and compared to analyse HARW aircraft. The empirical methods were coded in Excel spreadsheets and compared with AVL software. The empirical aerodynamic equations were later corrected according to the results obtained from the validation process. Sun proposed that the aerodynamic module was suitable for the conceptual design of HARW aircraft with reasonable accuracy.

2.11.4 Design Synthesis

Reiser (2014) studied the effect of aircraft layout on UHARW and SBW aircraft performance. In addition to the modules developed by other authors, Reiser (2014) also utilised and evaluated the existing empirical methods of mass, aerodynamic analysis, mission analysis, control & stability.

The author compared the results of the existing empirical MA methods with data from Boeing 737-800 (Raymer *et al.*, 2011) and the AT-13 SBW aircraft designed at Cranfield University (Smith, 2013). The best empirical method for MA of each aircraft component was investigated and selected for the developed semi-empirical conceptual design synthesis tool.

The errors of the best-performing empirical MA method for wing group mass of the cantilever Boeing 737-800 and SBW AT-13 were 15% and 22%, respectively. However, the errors of operational empty masses were 3.12% and 4.94% for the cantilever Boeing 737-800 and SBW AT-13, respectively. It should also be noted that data from Boeing 737-800 (Raymer *et al.*, 2011) and AT-13 (Smith, 2013) used for the validation were based on other estimation methods. The validation of the methods with real aircraft data should be conducted. Considering the high portion of the wing group mass in the total aircraft mass, the evaluated error levels are likely to cause uncertainties for SBW and cantilever wings.

Apart from wing MA, other significant errors occurred in the propulsion ME of AT-13 due to the high bypass ratio nature of the new engine design. These results proved the necessity of novel wing and propulsion MA methods for SBW aircraft.

Reiser (2014) also studied the CG balance of AT-13 and Boeing 737-800 aircraft for the longitudinal static stability, fuel consumption, and mass using the handbook methods. It was shown that the CG balance of SBW can be achieved with conventional approaches.

2.12 High Aspect Ratio Cantilever Wings

This section reviews HARW aircraft studies conducted at Cranfield University and other studies on using advanced materials in HARW aircraft. An aeroelastic study for a short-range HARW aircraft design study has also been reviewed. The studies reviewed in this section can also be considered for the design of HAR SBW And TBW aircraft in future studies.

2.12.1 Research at Cranfield University

In a study conducted by Pontillo et al. (2018), a static and dynamic scaling methodology was introduced to facilitate the analysis and Multidisciplinary Design Optimization (MDO) of HARW aircraft. Their research aimed to reduce a full-scale aircraft to a cantilever beam-size model for wind tunnel testing while maintaining accurate structural, aerodynamic, aeroelastic, and flight dynamic characteristics. The process employed Cranfield University's in-house Beam Reduction and Dynamic Scaling (BeaRDS) Program.

This study split into Phase I and Phase II, outlining details of Phase I. The first phase centred on identifying and rectifying potential issues in both experimental and scaling procedures for application in Phase II. Notably, a 12% error was detected in the deflection analysis of the scaled model compared to the full-sized one. The paper also shared insights from Phase I to inform the upcoming Phase II procedures.

Hayes *et al.* (2019) analysed the aerodynamics and static/dynamic behaviours of an XB-2 short-haul HARW aircraft with a wingspan of 48m and an AR of 18.8. They used the minimal exergy destruction principle to optimise air circulation around the wing, resulting in bell or elliptical shapes for unconstrained and constrained spans, respectively, according to Prandtl's lift distribution. The study compared results to a conventional short-haul aircraft (AR 9.5, span 34m). Based on thermodynamic principles, the exergy method enhanced Greene's minimal entropy-generating lift distribution. The authors also sought to validate Greene's work via Prandtl's method comparison. Their findings indicated that, given the same take-off gross weight (TOGW) and specifications, XB-2 consumed less fuel. BeaR code was utilized for ME.

2.12.2 Aeroelastic Constraints

CALFUNOPT was used for the conceptual design study of HARW matters and the static and dynamic aeroelastic constraints by Butler *et al.* (1995). The study aimed to improve the existing CALFUNOPT programme by introducing a well-known V-g method for dynamic aeroelastic analysis. The existing Theodorsen method in the code was also used for more precise analysis. The combination and selection procedure of any of these two methods was implemented in the code. Strip theory was used for unsteady aerodynamic loads. The minimum weight was the optimisation objective for a HAR composite wing.

A case study was run for a 40-seat regional aircraft with a maximum TOGW of 14.500 kg. Three main observations were obtained as follows: first, the optimisation causes wash-in coupling of material that raises the flutter speed; second, the elastic axis was relocated onward to reduce the coupling between aerodynamic and structural response; third, to prevent the flutter the difference between the first natural frequencies of bending and torsional tend to increase. However, the minimal effect of the engine position on the design study was attributed to the unsuitable sensitivity weightings.

2.12.3 Advanced Materials

Martins et al. (2014) conducted a study comparing the impact of various materials, including aluminium, carbon-fibre reinforced composites, and hypothetical carbon nanotube composites, on the performance of HARW aircraft. The optimization objectives were fuel consumption and Take-off Gross Weight (TOGW) reduction.

They used intermediate and high-fidelity tools to develop a static aeroelastic model. Aerodynamic analyses were conducted with a modified panel method (medium-fidelity) and a RANS CFD solver (high-fidelity). The structural analysis utilised a high-fidelity Finite Element Analysis (FEA) model. Both tools were coupled for aeroelastic analysis, and their derivatives were used for adjoint optimization in addition to gradient-based methods for exploring extensive design spaces.

Unexpectedly, the hypothetical nanotube composite underperformed compared to the conventional composite. Conventional and hypothetical composites reduced 7.7% and 5.2% in the quest for minimum fuel burn, respectively. Regarding minimum TOGW optimization, these materials provided 8.4% and 5.6% fuel reduction. All three materials displayed the same trends: optimal fuel consumption was achieved with longer, heavier, thinner, more elastic, and lightly loaded wings.

However, they noted the potential bias in the results due to a thickness constraint for the hypothetical composite identical to the conventional one, potentially skewing the comparative performances of these materials.

2.13 Gaps in the Literature

Across the literature, gaps exist in the understanding, research, and design methodologies of SBW and TBW aircraft. These gaps can broadly be categorised into general gaps and those that are directly addressed within the scope of this thesis.

Gaps Addressed in this Thesis:

- There is a need for innovative design solutions to reduce drag at strut-wing and strut-fuselage interfaces and to prevent strut buckling during negative loading (-2g) and taxi-bump scenarios while avoiding excessive strut weight.
- Direct solutions to the redundant structure problem built on logical assumptions and simplifications can be proposed, considering the unlimited number of wing-mounted concentrated loads.
- Current ME tools cannot evaluate the jury's impact and offset design parameters with enough accuracy and sensitivity on the overall design, limiting design exploration.
- Structural sizing and mass estimation methods that consider the unique moderate of HAR SBW and TBW configuration shapes and can model cutting-edge materials and emerging propulsion technologies, such as distributed propulsion and electric hydrogen-powered aircraft, are currently absent.
- There is no clarity for the reduced number of critical loading cases that drive the structural sizing of the novel wings in the literature.
- The effect of the distributed propulsion on the structural sizing and mass of the novel SBW and TBW configurations has not been sufficiently addressed.

This thesis aims to address these gaps and contribute to the field by developing a versatile, efficient, and sensitive MM for wing SSaME that surpasses existing methods' traditional limitations, offering greater accuracy and design space for novel aircraft configurations.

2.14 Requirements of the Mathematical Model

Given these gaps, the mathematical model developed in this thesis will aim to:

- Model existing aircraft configurations while assessing off-design performance.
- Study the sensitivity of performance relative to different design parameters.
- Scrutinize the impacts of cutting-edge materials or engine technology on wing structural design.
- Evaluate novel designs, particularly SBW configurations with emerging propulsion technologies.
- Generate any number of realistic loading cases, including gust, symmetric, rolling, and combined manoeuvres.

- Be compatible with reduced-order computational fluid dynamics (CFD) and other design methods.
- Generate any wing planform.
- Be sensitive to aircraft design parameters significantly affecting strut, truss, offset, wing box, and wing group mass.
- Maintain simplicity and speed, with potential for integration into multi-fidelity and multi-disciplinary optimization (MDO) tools.

The new methodology will aim to balance fidelity with computational efficiency and acknowledge its limitations in accuracy and design space due to empirical estimations and structural idealisations. The goal is to deliver a comprehensive, sensitive, and efficient tool for SDaME studies in conceptual aircraft design.

3 AIRCRAFT GEOMETRIC MODEL

A novel 3D aircraft geometric model has been developed for the conceptual design of innovative configurations, encompassing key structural components such as the fuselage, wing, strut, jury, offset, tail, canard, fin, and concentrated loads, which may include engines, landing gear, fuel drop tanks, and other elements. As the focus of the research is on the SDaME of the wing box, a more sophisticated geometric model is created for the lifting surfaces (LSs). In contrast, the fuselage geometric model exhibits a comparatively lower level of fidelity, as it is primarily required for empirical equations and calculating the mass moment of inertia (MMoI) for the entire aircraft. The external concentrated loads' geometry is assumed to be a simple cylinder.

Table 8 Reference points of aircraft components.

Component Name	Reference Point
Global Aircraft Coordinate System	The nose of the fuselage
Fuselage	The nose of the fuselage
Wing (LSs)	The joint of LE and wing centre line
Concentrated Loads	CG of the components
Tail	Same with the wing
Fin	Same with the wing

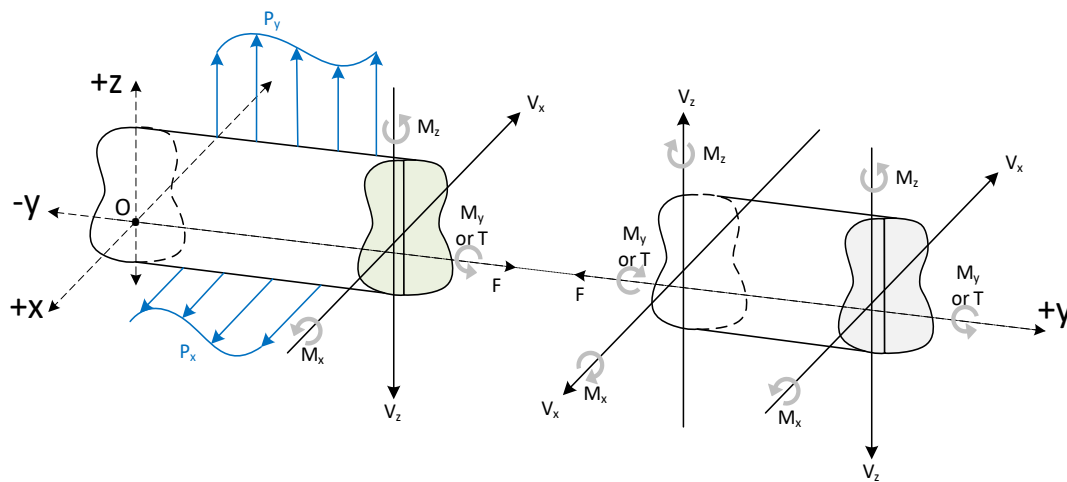


Figure 14 Sign convention used for the developed methodology.

Each component possesses its coordinate system, with the origin situated at the reference point of each respective component, as detailed in Table 8. These components

can be positioned within the global aircraft coordinate system to generate a comprehensive 3D representation of the aircraft.

The sign conventions for the geometric model, loading actions, and structural analysis are presented in Figure 14 with the positive directions. Unless otherwise specified, this sign convention will be used in the following chapters.

3.1 Cantilever Wing Geometric Description

The novel 3D wing geometric model, depicted in Figure 15 and Figure 16, is versatile and can be employed for a cantilever wing, the upper and lower wings of a JW, the wing and strut of an SBW, as well as the fin, tail, and canard of any aircraft. The model allows for variable wing bay and section count, distribution shapes, chord, thickness, sweep angle, dihedral or anhedral, spar positions, wing box shape, stringer count, and spar cap area to stringer area ratio. This flexibility enables the generation of any wing planform with unlimited numbers of unique wing sections.

To produce the 2D top and front views of the wing planform, users must define the values of N_{Sec} , $\varepsilon_{fs,j}$, $\varepsilon_{rs,j}$, c_j , τ_j , θ_j , β_j . When a lower-level fidelity is necessary during the initial design stage, these values will be calculated using linear interpolation between the nearest known values or default values if they are not assigned at all sections. Equation (3-1) presents the calculation for wing chord length at any wing section. Unless otherwise specified, this linear interpolation equation will be utilised for similar linear interpolation calculations.

$$c_j = (c_{j-1} - c_{j+1}) \left(\frac{\Lambda_{j+1} - \Lambda_j}{\Lambda_{j+1} - \Lambda_{j-1}} \right) + c_{j+1} \quad (3-1)$$

3.1.1 Coordinates of the Wing

First wing bay length ($b_{Bay,j}$) should be calculated with b_{Wing} and $N_{Sec,Wing}$. Assuming that the wing sections are distributed equally along the wing for simplicity, $b_{Bay,j}$ can be calculated with Eq.(3-2).

$$b_{Bay,j} = \frac{b_{Wing}}{2} * (N_{Sec,Wing} - 1) \quad (3-2)$$

In Eq.(3-2) $b_{Bay,j}$ is the distance between the wing sections along the global y-axis; however, the structural span length of the bays ($b_{Bay,STj}$) changes with the change of

sweep angle of the wing box reference line, $\theta_{RL,i}$. Hence, $b_{Bay,STj}$ can be calculated by Eq. (3-3)

$$b_{Bay,STj} = b_{Bay,j} / \cos \theta_{RL,j} \quad (3-3)$$

In Eq. (3-3), $\cos \theta_{RL,i}$ can be computed from $\theta_{L,i}$ which is the known design parameter of the wing. In addition to that, a general equation can be generated to compute the sweep angle, $\theta_{x,i}$, at any ND chord-wise position as fraction wing chord, $\varepsilon_{x,j}$, at a wing-bay using Eq. (3-4).

$$\theta_{x,j} = \arctan \left((\tan \theta_{LE,j} b_{Bay,j} + c_{j+1} * \varepsilon_{x,j} - c_j * \varepsilon_{x,j}) / b_{Bay,STj} \right) \quad (3-4)$$

3.1.1.1 Wing Reference Line

The reference line changes the wing planform with the wing design parameters. It is located at the intersection of the wing leading edge (LE) in the x-y plane (Figure 15) and the middle of the upper and lower surfaces in the y-z plane (Figure 16). Knowing the position of the wing reference points from Table 8, the coordinate of the wing reference line, $RF_j(x, y, z)$, in Figure 15 and Figure 16 can be calculated with Eqs. (3-5)-(3-7).

$$AF_j = b_{Bay,j} \times \tan (\beta_j) \quad (3-5)$$

$$AT_j = b_{Bay,j} \times \tan (\theta_{LE,j}) \quad (3-6)$$

$$R_j(x, y, z) = \left(\sum_0^j AT_j, y_j, \sum_0^j AF_j \right) \quad (3-7)$$

3.1.1.2 Top View

Assuming that the origin of the local axes of the wing is at the joint of the wing LE and the aircraft centreline, the coordinates of the wing planform in Figure 15 can be calculated with Eqs. (3-8)-(3-9) for the x-y plane.

$$WP_{LE,j}(x, y) = R_j(x, y) \quad (3-8)$$

$$WP_{TE,j}(x,y) = (WP_{LE,j}(x) + c_j, y_j) \quad (3-9)$$

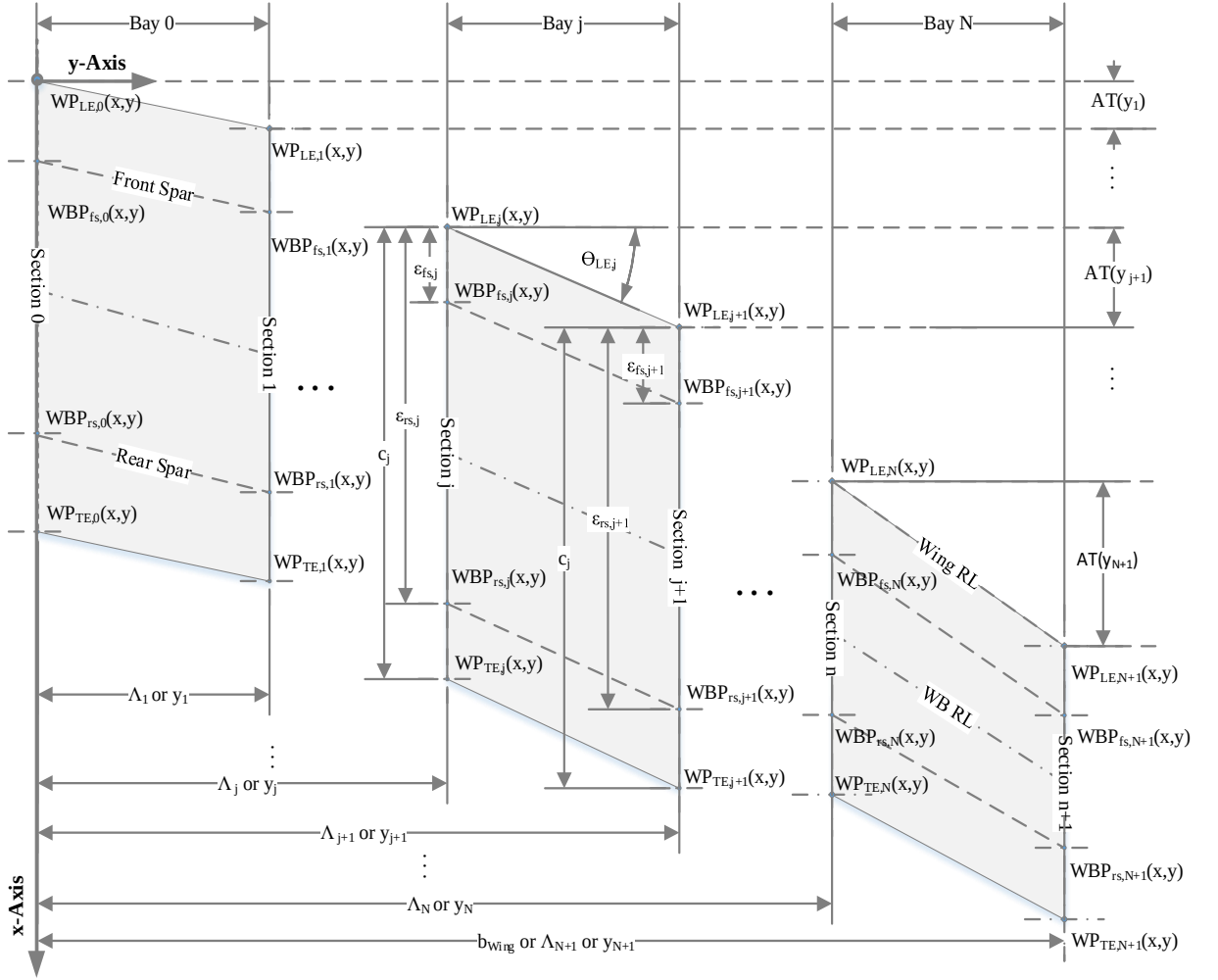


Figure 15 2D top view of new wing geometric model.

3.1.1.3 Front View

When the wing reference line in the y-z plane is known, the upper and lower surface coordinate points in Figure 16 can be calculated with Eqs. (3-10)-(3-11).

$$WP_{u,j}(y,z) = (y_j, R_j(z) + t_{wing,j}/2) \quad (3-10)$$

$$WP_{l,j}(y,z) = (y_j, WP_{u,j}(z) - t_{wing,j}) \quad (3-11)$$

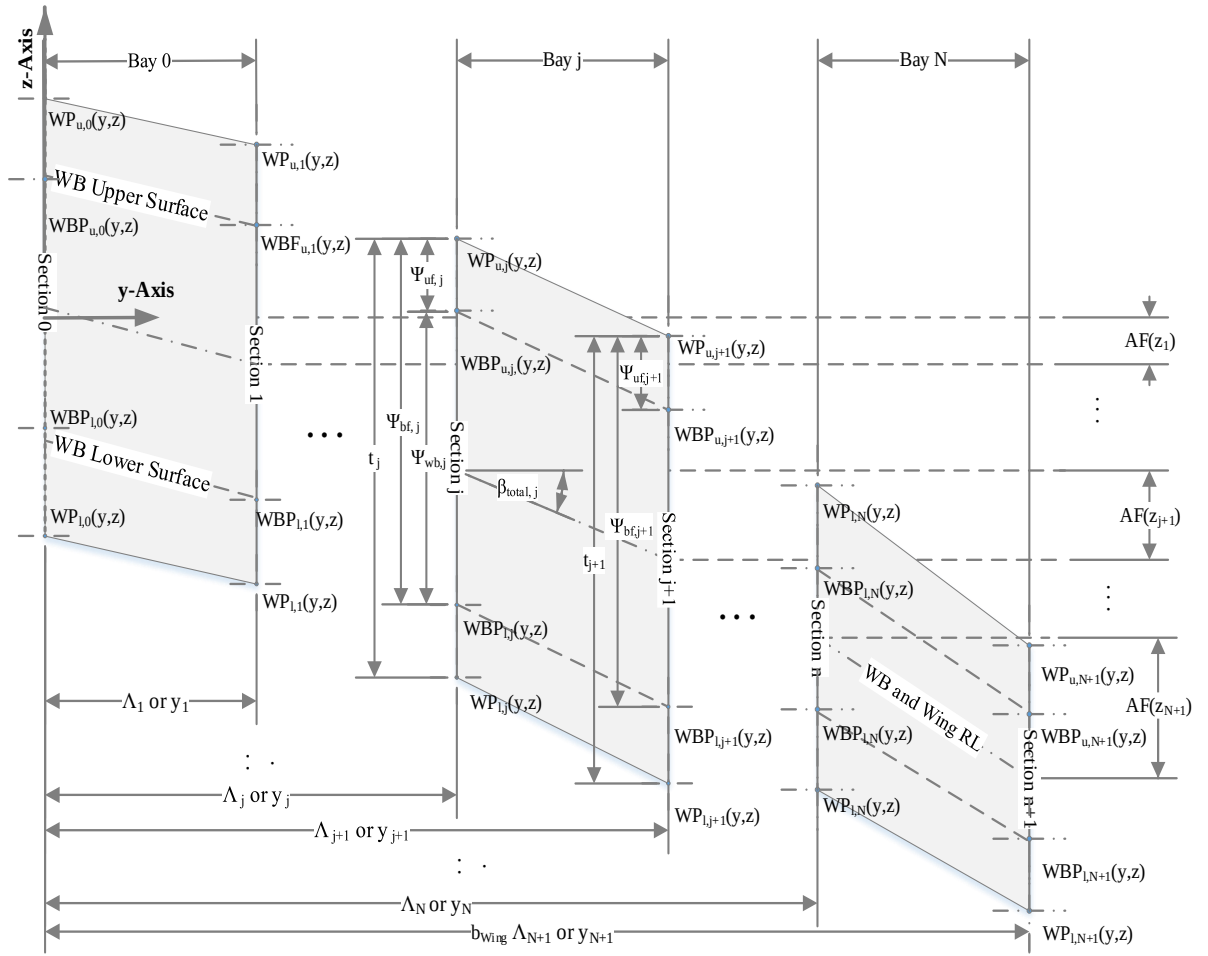


Figure 16 2D front view of wing geometric model.

3.2 Wing Box Models

In this study, two wing box models are introduced and used. The higher fidelity hexagonal model is used for the wing due to the complex loading actions and required higher fidelity stress analysis. A lower fidelity level rectangular model is chosen for the strut, jury, and offset boxes.

3.2.1 Irregular Hexagonal Box Model

A new, symmetric around two axes, wing box model with an irregular hexagonal shape is generated for this study following the findings from the previous studies in the field as discussed in Chapter-2 LITERATURE REVIEW. Unless it is called regular hexagonal, hexagonal means irregular hexagonal geometric shape in this thesis.

Structural idealization is used to reduce the computational time in the structural analysis of wing boxes in the conceptual or preliminary design stage. Subsonic or transonic aircraft with moderate or HARWs are usually designed with single-cell wing boxes; hence, an idealized single-cell wing box model is developed following the procedure (Megson, 2013). The idealization process is illustrated in Figure 17.

The following assumptions are made in the development of the new wing box model.

- Stringer counts are equal on the upper and lower surfaces of the wing box.
- The wing box is symmetric around two axes.
- The cross-sectional area of the spar flanges can be scaled to that of the stringer area using a ratio of SCR .
- Wing box edge height is a non-dimensional (ND) variable, $\Psi_{Edge,j}$, as a fraction of wing thickness. The thickest point of the idealized wing box is also an ND variable, Ψ_u , as a fraction of wing thickness.

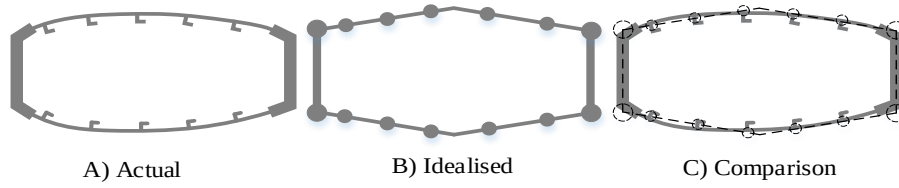


Figure 17 Illustration of the wing box idealization and comparison of actual and idealized WB section.

The wing box is generated with inputs of $\Psi_j, \varepsilon_j, \tau_j$ and N_{Boom} and the wing planform coordinates explained above. The wing box shape can be transformed from a diamond to a hexagon and a rectangular shape with the change of the $\Psi_{Edge,j}$ as can be seen in Figure 18.

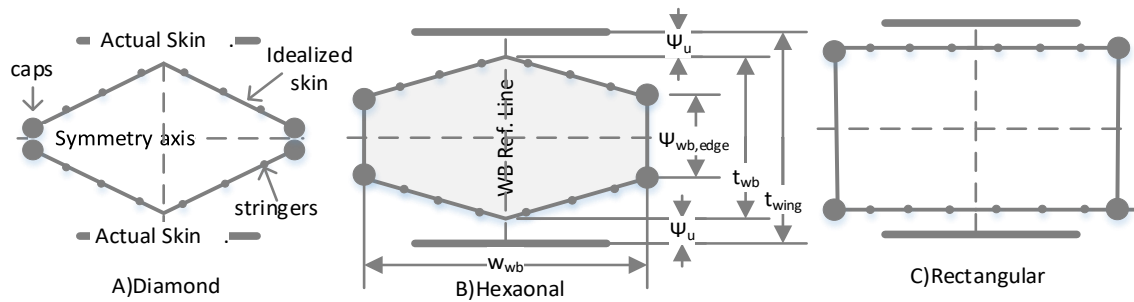


Figure 18 Geometric model representation of the wing box.

Wing and wing box section thicknesses are two different variables in the model. Wing box height ($t_{wb,j}$) is calculated from the $t_{wing,j}$ at local sections using Eq.(3-12).

$$t_{WB,j} = t_{wing,j} \times (1 - \Psi_u) \quad (3-12)$$

where $t_{wing,j}$ can be calculated with Eq.(3-13).

$$t_{wing,j} = c_j \times \tau_j \quad (3-13)$$

Finding $t_{WB,j}$, $t_{WB,Edge,j}$ can be calculated with the user input of $\Psi_{WB,Edge,j}$ to generate the final geometric shape of the wing box with Eq.(3-14).

$$t_{WB,Edge,j} = t_{WB,j} \times \Psi_{WB,Edge,j} \quad (3-14)$$

Finding $t_{WB,j}$, $t_{WB,Edge,j}$ can be calculated with the user input of $\Psi_{WB,Edge,j}$ to generate the final geometric shape of the wing box with Eq.(3-17).

3.2.2 Coordinates of the Wing Box

3.2.2.1.1 Top View

After computing the values of $WP_{LE,j}$, $WP_{TE,j}$, $WP_{u,j}$, $WP_{l,j}$ for the wing, the wing box coordinates of $WBP_{l,j}$, $WBP_{r,j}$, $WBP_{u,j}$, $WBP_{b,j}$ in Figure 15 and Figure 16 can be calculated as follows.

$$WBP_{fs,j}(x, y) = (WP_{LE,j}(x) + c_j \times \varepsilon_{fs,j}, y_j) \quad (3-15)$$

$$WBP_{rs,j}(x, y) = (WP_{LE,j}(x) + c_j \times \varepsilon_{rs,j}, y_j) \quad (3-16)$$

$WBP_{fs,j}$ and $WBP_{rs,j}$ are also the coordinate points of the wing spar at the x-y plane; hence the wing box width ($w_{WB,j}$) can be calculated Eq.(3-17).

$$w_{WB,j} = WBP_{fs,j}(x) - WBP_{rs,j}(x) \quad (3-17)$$

3.2.2.1.2 Front View

Recalling $R_j(x, y, z)$ from Eq. (3-7), wing box front view coordinate points in Figure 16 can be calculated with Eqs. (3-18), (3-19).

$$WBP_{u,j}(y, z) = (y_j, R_j(z) + t_{WB,j}/2) \quad (3-18)$$

$$WBP_{l,j}(y, z) = (y_j, R_j(z) - t_{WB,j}/2) \quad (3-19)$$

The wing box reference line is positioned at the intersection of the middle of the two spars in the x-y plane and the middle of the wing box's upper and lower surface. Hence the coordinates of the WB reference line can be computed with Eqs. (3-20).

$$R_{WB,j}(x, y, z) = \left(\frac{WBP_{rs,j}(x) + WBP_{fs,j}(x)}{2}, y_j, \frac{WBP_{u,j}(z) + WBP_{l,j}(z)}{2} \right) \quad (3-20)$$

3.2.3 Pros and Cons of Symmetric Wing Box Model

The symmetric around two axes wing box model, chosen for computational efficiency, aligns well with the selected aerodynamic models' fidelity level, and the empirical aeroelastic methods, ensuring a cohesive and practical approach. It was reviewed in Section 2.1.3.2 that the symmetric wing box model approach has been used in several mass estimation methods in the literature together with simpler wing box models of rectangular, I beam, and double plate.

The presented wing box model, one of the most detailed in the literature, consolidates stringers, spar caps, and webs, offering detailed representation while maintaining computational simplicity. With the symmetric wing box model, the direct computation of the size of those structural elements was possible without using an iterative solution which could significantly increase the computational time. The validity of this approach is shown in validation studies with acceptable error values which were the lowest compared to other existing mass estimation methods.

However, users are notified about its applicability to Ultra-High Aspect Ratio Wings, where wing box shear centre or flexural axis positions significantly impact aeroelastic characteristics. The wing mass estimation method is validated with HAR NASA SUGAR Aircraft; however, at this stage of the project due to the limited available data, it is not possible to test the effect of those assumptions on the mass estimation of UHARWs. Hence, future studies may explore more intricate models for these scenarios. As will be discussed in Chapters 10 and 12, the presented methods require more detailed validation in the mass estimation of UHARWs.

3.2.4 Locating the Stringers

Any number of stringer counts can be assigned as an input on the upper and lower surfaces by changing the value of N_{Boom} . The number of stringers on the upper and lower surfaces is equal due to the assumption of wing box symmetry along both axes. Stringers are distributed equally, crossing their reference line with the idealized wing box skin. The two spars of a single-cell wing box are consistent along the wing from the root to the wing tip, and they can be positioned using Eqs. (3-15) and (3-19).

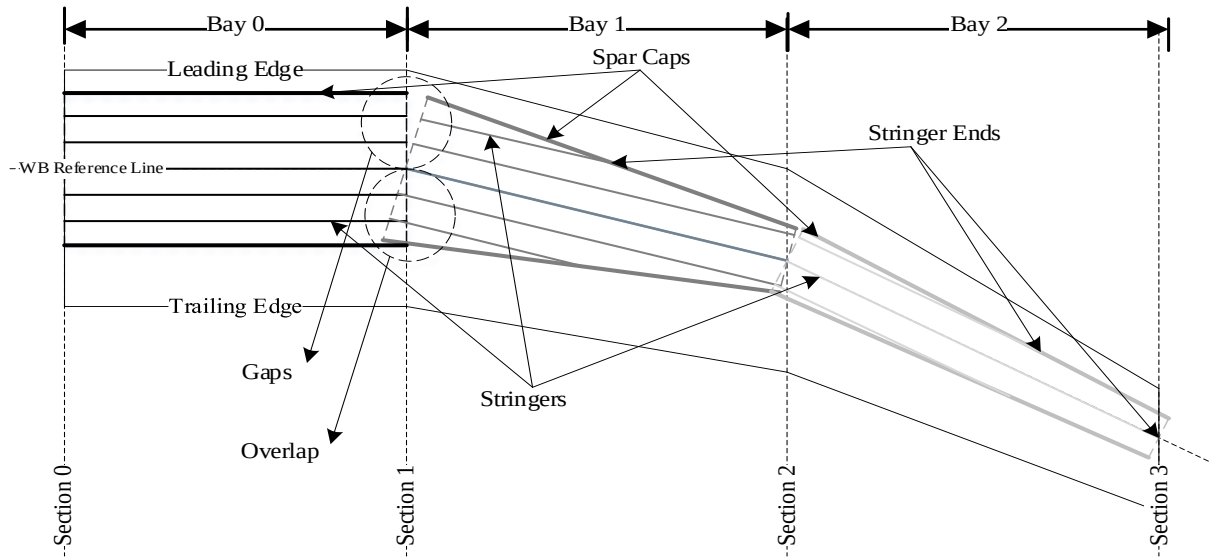


Figure 19 Location of the stringers.

The stringers are assumed to be positioned parallel to the wing box reference line with a constant pitch of $d_{stringer}$ which can be computed with Eq.(3-21). The stringers are cut when they come into contact with the spars; hence the stringer counts are reduced towards the wing tip depending on the wing taper ratio, as shown in Figure 19. Moreover, the gaps and overlaps, illustrated in Figure 19, occur when the section sweep angles change, and their effects are neglected as they are assumed to cause an acceptable level of error in the structural analysis.

$$d_{stringer} = w_{WB,Root} / (N_{Boom} / 2 + 1) \quad (3-21)$$

3.2.5 Rectangular Box Model

Considering the trade-off between the computational time and the model accuracy, the load-carrying structures of the strut, the jury, and the offset are modelled as rectangular boxes, as illustrated in Figure 20. Although the rectangular box model is simpler than the

idealized hexagonal model used for the wing box in this study, it is expected to provide enough accuracy for the components it represents. The rectangular box model requires less computational time and can easily be coded for automatic sizing and ME. The sizing of strut and offset will be explained in the next chapters using the rectangular box model and sizing constraints.

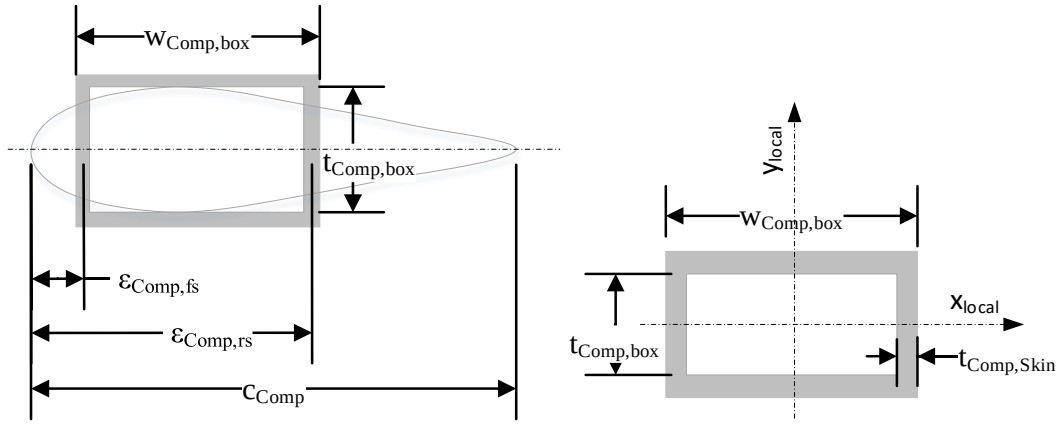


Figure 20 Strut and offset box model.

As the derivation of coordinate points of rectangular box models without mass booms is relatively easy, especially for uniform beams, their detailed derivations are not repeated in this section, and they can be derived following the similar steps of the hexagonal box model presented in the previous section.

Furthermore, if more detailed results are required, an idealised hexagonal box model can also be used for the truss components following the procedure presented in the previous sections. However, in this study, only the rectangular box model is used for the ME of the strut, jury and offset load-carrying structures.

3.3 Strut- and Truss-Braced Wings Geometric Models

This section describes strut, jury, and offset geometric models. Considering the fidelity level of the stress analysis and sizing, a simpler geometric model is generated for the strut, jury and offset compared to the wing. However, if required wing geometric model can also be employed for higher fidelity modelling of truss components.

3.3.1 Strut Geometric Model

The strut geometric model is created considering its design variables, loads acting on it and required fidelity level to size it accurately.

The input parameters for the strut are ND strut position along the wing as a fraction of half wing span length, λ_{Strut} , ND chordwise position of the strut as a fraction of the local wing section chord, ε_{Strut} , strut chord ratio to the wing local chord where the strut is attached, $\tau_{StrutToWing}$, strut thickness to chord ratio, τ_{Strut} , strut sweep angle (From the top view, CCW direction is positive), θ_{Strut} , the strut root attachment point distance to the aircraft centre line in the y-axis, $y_{Strut,Root}$ and in z-axis $z_{Strut,Root}$. Unless required, $y_{Strut,Root}$, $z_{Strut,Root}$, can be set as default values of zero and half of the fuselage diameter, respectively.

The attachment of the strut to the wing is illustrated in Figure 21. When ε_{Strut} is set to the zero, the strut is attached to the wing LE. Other parameters are used for the structural analysis.

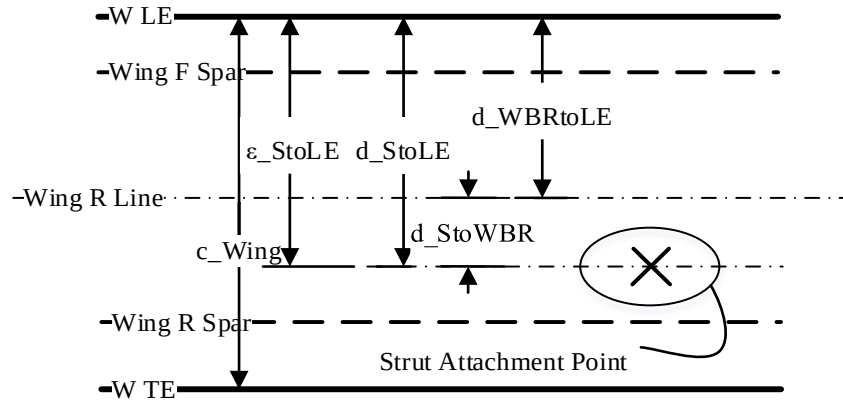


Figure 21 Strut-wing attachment top view.

Figure 22 and Figure 23 illustrate the details of the top and the front views of the strut and the offset geometric models. Without an offset, the strut is directly connected to the wing. The offset changes the coordinates of the strut tip and the root in the x and z axes, depending on the offset height and sweep angle.

The strut box is represented by the rectangular model, as explained previously, and the default value of the forward ($\xi_{Strut,fs}$) and the rear spars ($\xi_{Strut,rs}$) positions of the strut box are set to ND positions of 0.15 and 0.65 as a fraction of the strut chord; they can be seen in Figure 22.

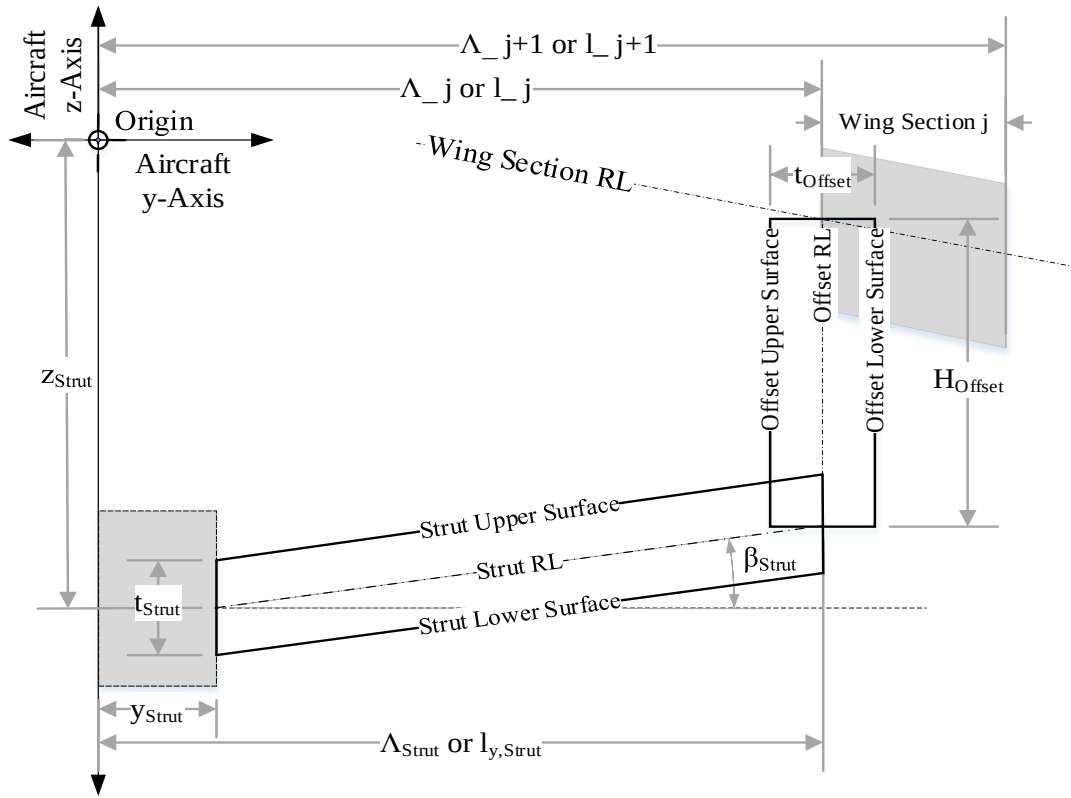


Figure 23 2D front view of the SBW geometric model.

3.3.2 Offset Geometric Model

Design variables shown in Figure 24 are utilized to create the offset geometry, and they are offset thickness to chord ratio (τ_{Offset}), offset chord ratio to the wing root chord ($\tau_{OffsetWing}$), offset height in global z-axis direction (H_{Offset}), and offset angle around the global y-axis (θ_{Offset}) (From side-view, CW is positive).

The rectangular model is used for the offset box. The default values of the ND front ($\xi_{Offset,fs}$) and rear spar ($\xi_{Offset,rs}$) offset positions as a fraction of the offset chord are set to 0.15 and 0.65, respectively. These values can also be set as inputs.

In the presence of the offset, the coordinates of the tip of the offset are computed using Λ_{Strut} and the same procedure to find the strut tip coordinates will be applied. The root of the offset is where it is connected to the strut, and it can be found with simple trigonometric relations using H_{Offset} and θ_{Offset} . It should be noted that in the presence of an offset, the coordinates of the strut tip will be equal to the coordinates of the root of the offset.

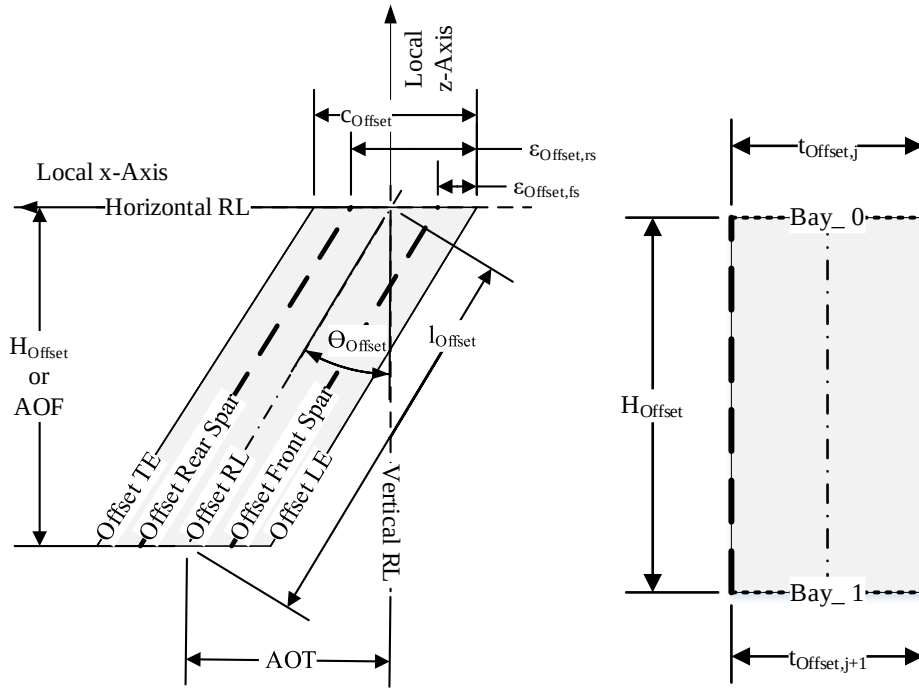


Figure 24 2D side and front views of the offset.

After computing the offset's tip and root coordinate points, an offset reference line can be generated between these points. Similar to the strut, 3D offset geometry can be generated using the offset design variables.

3.3.3 Jury Geometric Model

The juries are located between the strut and wing reference lines. It is observed in the literature that the juries can be designed either normal to the wing reference line (jury-1) or normal to the strut reference line (jury-2), as illustrated in Figure 25. Hence, these two types of jury directions are modelled and studied in this study.

Each jury has four design variables which are jury direction (either normal to the strut or the wing), jury chord ratio to that of the strut $\tau_{Jury,Strut}$, jury thickness to chord ratio $t_{Jury,j}/c_{Jury,j}$, and ND jury position in the y-axis as a fraction of strut length $\tau_{Jury-j,Pos}$. It should be noted that the strut root can be moved in the z-axis direction, hence if $\tau_{Jury-j,Pos}$, for example, if chosen zero, the root of the jury will be positioned at the strut root. Jury ND position along the wing as a fraction of wing half wingspan $\lambda_{Jury-j,Root}$; hence it can be calculated from $\tau_{Jury-j,Pos}$ considering the strut root coordinates.

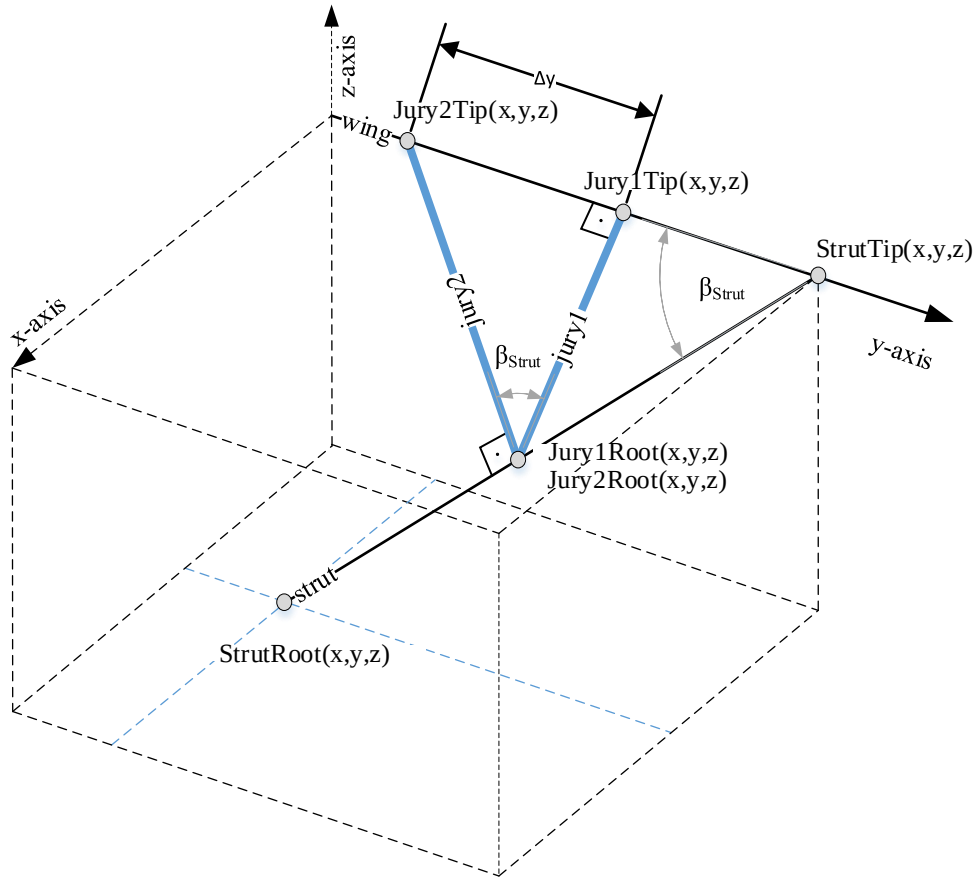


Figure 25 Jury directions are normal to the wing reference line (jury-1) or the strut reference line (jury-2).

If $\Delta_{Jury-j,Root}$ is calculated from the design variable $\tau_{Jury-j,Pos}$, the jury root position at the global y-axis can be calculated. If $y_{Jury-j,Root}$ is known, $x_{Jury-j,Root}$ and $z_{Jury-j,Root}$ can be found with geometric relations as formulated in Eq. (3-22), considering that the jury root is connected to the strut reference line.

$$\begin{aligned}
 x_{Jury-j,Root} &= x_{Strut,Root} - \frac{y_{Jury-j,Root} - y_{Strut,Root}}{y_{Strut,Tip} - y_{Strut,Root}} (x_{Strut,Root} - x_{Strut,Tip}) \\
 z_{Jury-j,Root} &= z_{Strut,Root} + \frac{y_{Jury-j,Root} - y_{Strut,Root}}{y_{Strut,Tip} - y_{Strut,Root}} (z_{Strut,Tip} - z_{Strut,Root})
 \end{aligned} \quad (3-22)$$

For a jury parallel to the global z-axis or normal to the wing without swept, $y_{Jury-1,Root}$ is equal to $y_{Jury-1,Tip}$. The jury tip will have the same coordinates point with the wing reference line at $y_{Jury-j,Tip}$, as the jury tip is connected to the wing reference line. Hence

$x_{Jury-j,Tip}$ and $z_{Jury-j,Tip}$ can be found by searching the coordinate points of wing reference line coordinates to match with $y_{Jury-j,Tip}$.

The root coordinate points of jury-2 (normal to the strut reference line) in Figure 25 have the same root coordinates as jury-1. It can be seen that the coordinate of the tip of jury-2 can be calculated using Eq.(3-23).

$$y_{Jury-2,Tip} = y_{Jury-1,Tip} + \Delta y \quad (3-23)$$

Where Δy can be computed using Eq.(3-24) and $\cos\beta_{Strut}$ in. Eq.(3-24) can be calculated using Eq.(5-57). Moreover, l_{Jury-1} can simply be calculated with the tip and root coordinates of the jury-1.

$$\Delta y = \frac{l_{Jury-1}}{\cos\beta_{Strut}} \quad (3-24)$$

After finding $y_{Jury-2,Tip}$, similar to jury-1, the tip coordinates of jury-2 can be found by searching the connection point among the wing reference line coordinates.

It should be noted that the geometric relations given for jury 2 are derived for a straight wing. Therefore, when the wing sweep or dihedral angles are changed, the jury-2 angle to the strut will slightly change and not be normal to the strut reference line. This small effect is neglected in the calculations.

After computing the tip and root coordinate points of the juries, a jury reference line can be generated between these two points, and a 3D view of the juries with their spar positions and airfoil shapes can be generated with the jury design variables following a similar procedure to that of the offset and the strut.

3.4 Concentrated Loads

For transport aircraft, landing gears, engines and fuel drop tanks are considered concentrated loads, and their geometry is assumed to be cylinders. An unlimited number of concentrated loads might be defined depending on the flight mission of the air vehicle. Except for the thrust of the engine, the concentrated loads are caused by inertia.

For the simplification of the structural calculations, the concentrated loads are assumed to be positioned only at the sections of the LSs using ND parameters of $\Psi_{Con,j,k}$ in z - axis, $\varepsilon_{Con,j,k}$ in x - axis, and $\Lambda_{Con,j,k}$ in y - axis relevant to the wing reference line and the dimensions of the wing local sections with Eq.(3-25). The ND parameters can be

defined by the designer or be optimized by the tool, and the examples of their optimisation will be illustrated in the following chapters.

$$\begin{aligned} Con_{j,k}(x, y, z) = & (R_j(x) + \varepsilon_{Con,j,k} \times c_j, \quad A_{Con,j,k} * \frac{b}{2}, \\ & R_j(z) + \Psi_{Con,j,k} \times t_{Wing,j}) \end{aligned} \quad (3-25)$$

3.5 Box Wing Geometric Model

Although the box-wing aircraft configurations are not directly covered in this thesis, the geometric models introduced in the previous sections are sufficient for modelling box-wing aircraft. If a higher fidelity level is required for both wings of the configuration, the strut can be modelled using the wing geometric model introduced.

This study will not cover the box wing's structural analysis and sizing process. The developed method in this study is not validated with box-wing aircraft data. However, in future research, these concepts can be studied using the fundamentals presented in this thesis.

3.6 Tailplane Geometric Model

The wing geometric model can create a horizontal tail and fin geometry. Depending on the complexity of the geometries, section numbers can be increased. In this study, two sections and one bay geometric model are created for the horizontal tail, and the fin using the method explained in Section 3.1 (Cantilever Wing Geometric Description).

3.7 Fuselage Geometric Model

A simple fuselage geometric model in Figure 26 is developed for the project. The fidelity level of the model is chosen due to the project's requirements. The fuselage model is primarily required for the simple calculations of mass moment inertia of the entire aircraft, and it can also be used for visual verification of the design and generating data for empirical fuselage ME methods and other sub-modules in an MDO environment.

The width and length of the fuselage can be defined by D_F and l_F , respectively. The nose shape and the tail cone are generated by the tool with default ND parameters as a fraction of fuselage diameter (ξ_i) and length (ζ_i) using 17 coordinate points ($FP_i(x, y)$) for the top view and 38 points for the front view. If required, the fuselage shape can be modified by changing the position of those design points.

Table 9 ND default parameters to create fuselage top view.

Section Name	Locations (i)	$FP_i(y)$	$FP_i(x)$
Aircraft Reference point	0	0.00	0.00
	1	$0.25D_F$	$0.08\zeta_{ND\text{Depth}}L_F$
Half Nose	2	$0.50D_F$	$0.20\zeta_{ND\text{Depth}}L_F$
	3	$0.750D_F$	$0.40\zeta_{ND\text{Depth}}L_F$
	4	$0.95D_F$	$0.70\zeta_{ND\text{Depth}}L_F$
	5	$1.00D_F$	$1.00\zeta_{ND\text{Depth}}L_F$
	6	$1.000D_F$	$(1-\zeta_{TC\text{Depth}})L_F$
Half Tail Cone and its Mirroring	7	$0.10D_F$	L_F
	8	0.00	L_F
	9	$0.10D_F$	L_F
	10	$1.00D_F$	$(1-\zeta_{TC\text{Depth}})L_F$
Mirroring the Half Nose	11	$1.00D_F$	$1.00\zeta_{ND\text{Depth}}L_F$
	12	$0.95D_F$	$0.70\zeta_{ND\text{Depth}}L_F$
	13	$0.75D_F$	$0.40\zeta_{ND\text{Depth}}L_F$
	14	$0.50D_F$	$0.20\zeta_{ND\text{Depth}}L_F$
	15	$0.25D_F$	$0.08\zeta_{ND\text{Depth}}L_F$

The ND parameters for the width variation of the fuselage can be seen in Table 9. For simplicity, it was assumed that the geometric shape of the front view of the fuselage is a circle; hence the fuselage shape front view is generated using a well-known circle equation with an input value of D_f .

3.8 Assembling the Main Components

All the main components, including the wing, tail and fin, can be located on the global axis of the entire aircraft using the ND parameters of ζ in the x-axis and ξ in both the y and z-axes, as seen from the illustration in Figure 26.

The origin of the global coordinate system is chosen at the nose of the aircraft. The position of the reference point of each component relevant to the global reference point should be defined with $\zeta_{WtN}, \zeta_{TtN}, \zeta_{FintN}, \zeta_{ND\text{Depth}}, \zeta_{TC\text{Depth}}$ in the x-axis to locate the components in the global coordinate system of the aircraft.

In the z-axis, the connection point of the wing, strut and fin can be changed. The engine location can also be defined in 3D dimensions in the global axis system for fuselage-mounted engines.

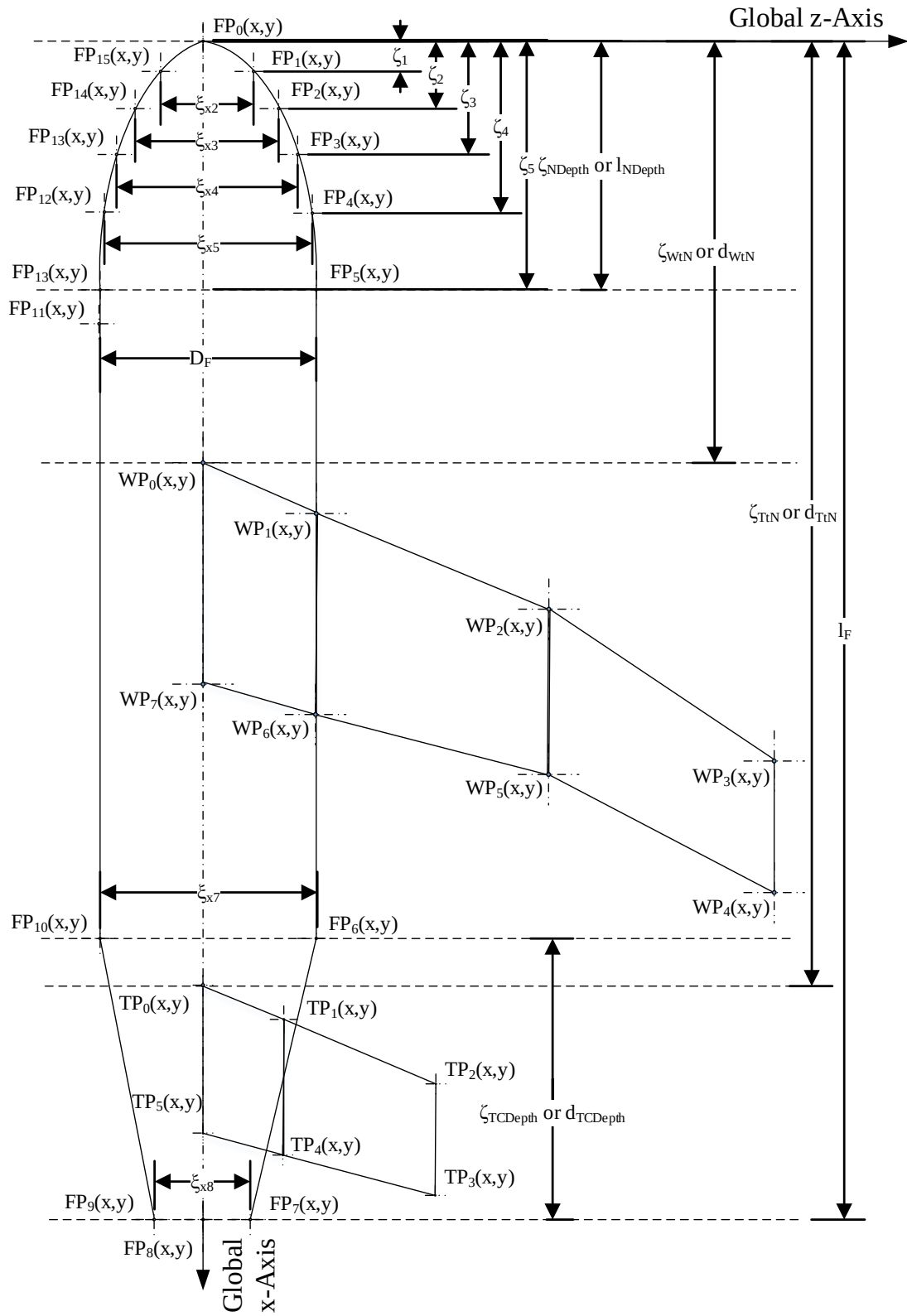


Figure 26 2D top view of fuselage geometric model.

3.9 2D and 3D Visualisations

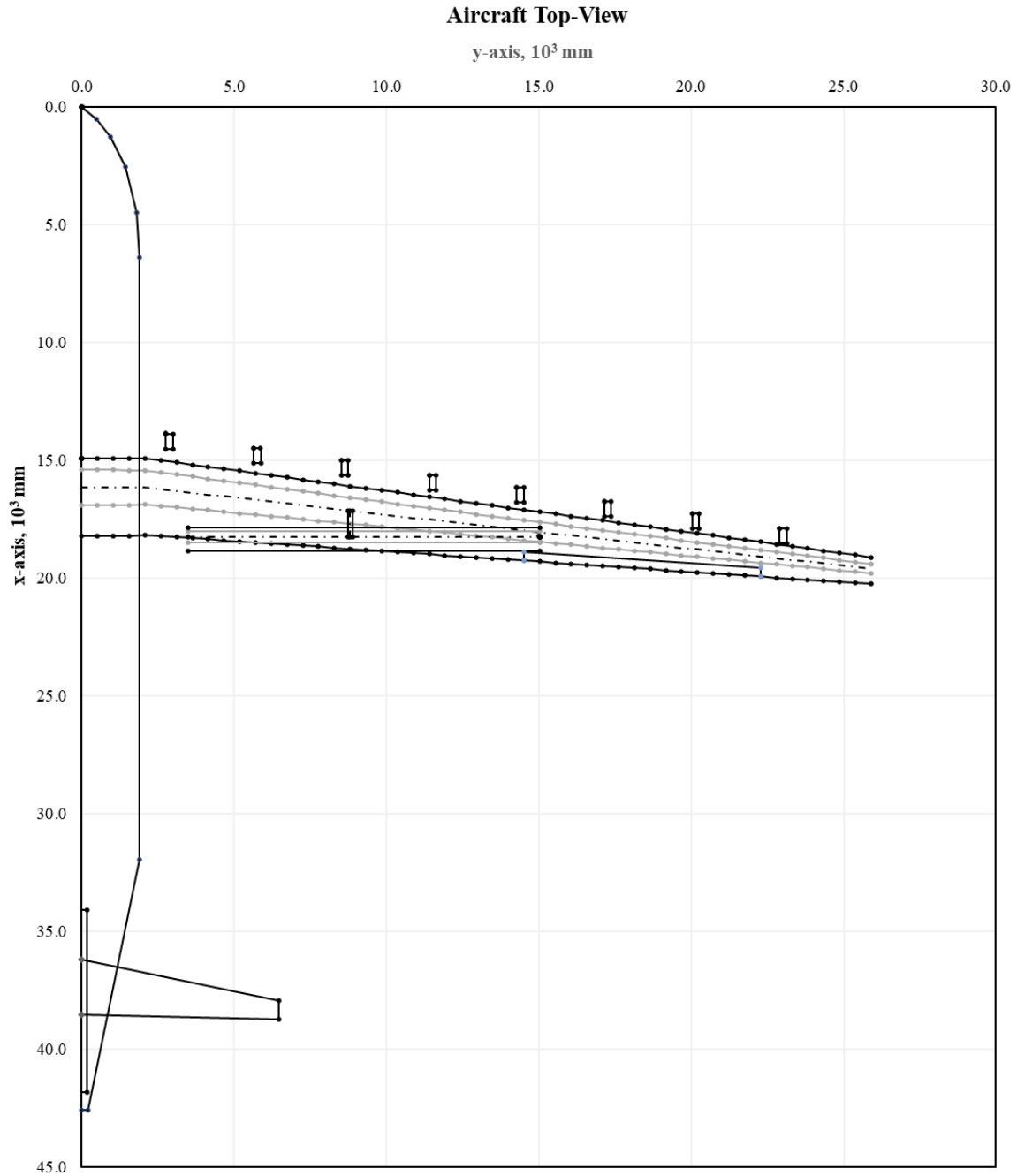


Figure 27 Fuselage 2D top view generated using the geometric relations introduced in this chapter.

This chapter introduces new geometric models of a novel aircraft's main components. Each component can be generated and located in a global coordinate system to create the entire aircraft. Therefore, 2D and 3D visualisation of the whole aircraft can be generated using the models presented in this chapter. A sample 3D view of an aircraft

with SBW and an offset and a 2D front and top view of a complete aircraft with TBW and distributed propulsions are illustrated in Figure 1, Figure 28 and Figure 27, respectively. This visualisation tool verifies the geometric formulas developed above and can be used for design.

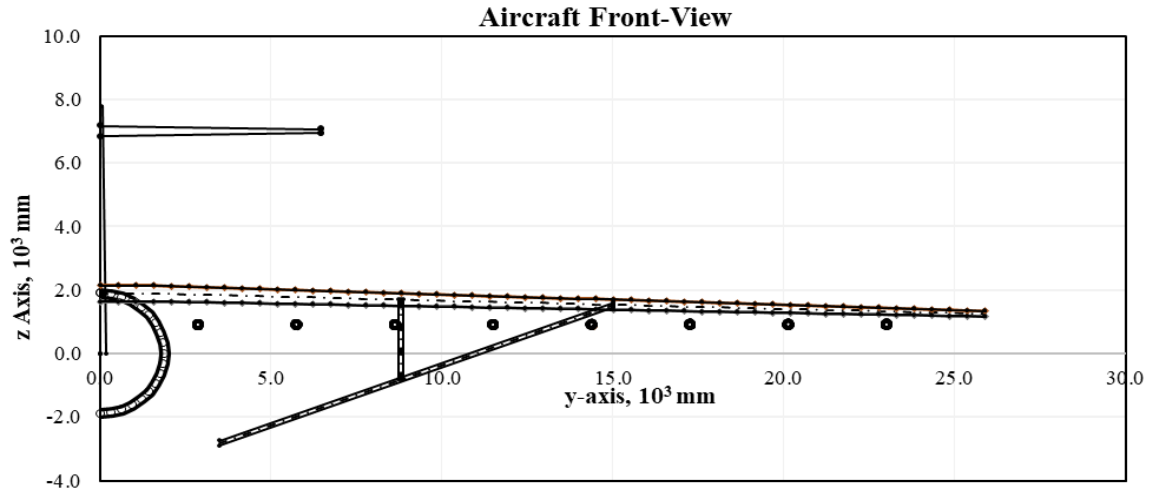


Figure 28 Fuselage 2D front view generated using the geometric relations introduced in this chapter.

4 LOADING ACTIONS

This chapter explores the loading cases crucial to developing an aircraft wing's SDaME within a mathematical model (MM). This analysis is vital for understanding the intricate interplay of forces influencing an aircraft's performance and efficiency during various flight phases. Aircraft are categorized based on missions, top-level aircraft requirements (TLAR), and the type of loads they are subjected to. It is essential to briefly distinguish between CS-23 (EASA, 2015) and CS-25 (EASA, 2020) certification requirements, as these frameworks dictate different design criteria and performance standards for diverse aircraft categories.

CS-23 requirements pertain to small aircraft with a maximum certified take-off weight not exceeding 19,000 lbs (8,618 kg) or a maximum seating capacity of 19 passengers, including general aviation, personal, and small business aircraft. In contrast, CS-25 requirements apply to transport category aircraft designed primarily for commercial air transportation, with a maximum certified take-off weight exceeding 19,000 lbs (8,618 kg) and a seating capacity greater than 19 passengers.

The flight envelope of an aircraft and the required loading cases for structural design are defined by airworthiness requirements, such as CS-23 and CS-25. This study follows these requirements for transport aircraft, emphasizing their continued relevance as vital guidelines for innovative configurations, including transport aircraft with TBWs, wings with distributed propulsion, and electric or hydrogen power technologies. The methods discussed in this chapter can be applied to the conceptual or preliminary design of other aircraft categories, demonstrating the versatility of this approach in adhering to airworthiness requirements.

4.1 Airworthiness Requirements

This project mainly focuses on large transport aircraft hence the application of CS-25 (EASA, 2020) requirements for loading actions will be given in more detail. CS-23 requirements can be generated with less complexity following CS-23 (EASA, 2015) and a similar computation process presented in this chapter.

Two sub-modules are developed to generate the loading cases addressed in CS.23 and CS.25. The modules automatically calculate design speeds and loading factors and generate flight envelopes. Aerodynamic, damping and inertia loads acting on the wing are calculated using these outputs.

Table 10 The parameters in the table are combined to generate the symmetric loading cases.

Direction	Altitudes (ft)	Fuels Capacity	Payloads Capacity	Speeds
Positive	0	100	100	V_B
Negative	10000	20	0	V_C
	20000	0		V_D
	30000			
	40000			

Table 11 The parameters in the table are combined to generate rolling and combined cases.

Rolling Direction	Rolling Cases	Manoeuvre in Rolling	Altitudes (ft)	Fuels	Payloads	Speeds
Rolling Up	Initiation	n0	0	100	100	V_A
Rolling Down	Steady	n2\3	40000	0	0	V_C
	Arrest					V_D

Table 12 The variable values changed with different cases.

Case Number	421	Load Case Type	Manoeuvring	Diving Velocity (EAS) (m/s)	122.4
Load Factor, n (g)	2.5	Manoeuvring In rolling Case	n2\3	$M_{Aircraft,case}$ (kg)	312,046.0
Fuel (%)	100	Manoeuvring Direction	Positive	p_{case} , rad/s	0
Payload (%)	0	Rolling Case Type	Initiation	$\bar{p}_{case}$, rad/s ²	-0.212
Altitude (ft)	44,996.7	Rolling Direction	Rolling Down	$\bar{Z}_{Aileron,case}$ (N)	265,976.0
Altitude (m)	13,715.0	Design Speed	V_C	Air Density (kg/m ³)	0.239
Altitude (ft)	44996.7	Case Velocity (TAS) (m/s)	256.7	Dynamic Pressure (N/m ²)	7,863.9

CS.23 and CS.25 require designing aeroplanes under different loading cases generated by the combination of different directions, altitudes, fuel capacities, payload capacities, and speed variations, as shown in Table 10. As a result, 195 symmetric loading cases are created. In addition to 195 cases, more cases arise from rolling conditions and the combination of rolling and symmetric manoeuvring. Hence a total of 483 cases are generated for CS-25 (EASA, 2020) requirements, including gust, manoeuvre, rolling and combined cases as the default setting of the tool. Rolling cases cover arrest,

aerodynamic damping, and steady cases in addition to different altitudes, fuel loads, and payloads, as shown in Table 11. Taxi bump and landing cases are also covered.

When a case is changed, the values in Table 12 are changed; therefore, any dependent parameters changed in the loading action and other sub-modules. Different loads occur in each flight case, and their calculation process is shown in the following sections.

The detailed derivations to compute the flight envelopes for symmetric and combined loading cases following CS-25 (EASA, 2020) certification requirements are presented in 12.2 Appendix A - Loading Actions.

4.1.1 Safety Factor

This section will explain the terminology in CS-25 (EASA, 2020) for safety factors and its application in this project.

Limit loads and ultimate loads (CS 25.301)

The stipulated strength requirements are expressed in limit loads (the maximum anticipated service loads) and ultimate loads (limit loads multiplied by the predetermined safety factors). Unless specified otherwise, the stated loads are considered limit loads.

Safety Factor (CS 25.303)

Unless stated otherwise, a safety factor of 1.5 should be applied to the designated limit load, regarded as external loads on the structure. When the loading condition is defined in terms of ultimate loads, a safety factor is not required unless specified differently.

Structural Integrity and Deformation CS 25.305

(a) The structure must have the capacity to withstand limited loads without incurring harmful permanent deformation. Deformation should not compromise the safe operation for any load up to the limit loads.

(b) The structure must endure ultimate loads without failure for a minimum duration of 3 seconds.

In the developed MM for wing structural sizing and mass estimation (SSaME) of transport aircraft, these requirements are followed while making certain assumptions.

4.1.2 Structural Sizing Criteria

Application for isotropic materials (metal alloys)

For aluminium alloy materials, the limit loads are applied as prescribed by the CS 25.305(a) criteria and yielding stress of the material for the sizing of the wing components, ensuring that harmful permanent deformation does not occur.

Application for quasi-isotropic materials (composites)

For composite materials, the ultimate load criterion (CS 25.305(a)) together with the failure stress of the material were used. However, due to the model's non-time-dependent and non-dynamic nature, the 3-second requirement could not be tested. Instead, the composites were sized using static analytical models based on failure stress.

4.2 Load Distribution Models

Load distribution models are required when applying the total limit or ultimate loads along the wing or any lifting surfaces. Although the total lift generated by the lifting surfaces or the weight of aircraft components can be estimated at the initial design stage, the distribution shape of those loads may not be known and should be assumed. Some simple distribution models, including triangular, trapezoidal, and elliptical, can provide acceptable accuracy depending on their application (Schrenk, 1940; JT, 2001; Sun, 2014; ÜNAY, 2015; Kamal *et al.*, 2019). The applications of these types of distributions are explained in this section.

4.2.1 Triangular Distribution

A triangular load distribution model, as presented in Figure 29, can be created with fundamental geometric relations. The load per unit length (N/mm) at the root, $P_{Tri,Root}$, and at the tip, $P_{Tri,Tip}$, of the structural component can be calculated with Eq. (4-1) for triangular load distribution.

$$\begin{aligned} P_{Tri,Tip} &= 0 \\ P_{Tri,Root} &= (2 \times F)/b \end{aligned} \tag{4-1}$$

where F can be any load distributed along the span of the LS hence, $P_{Tri,j}$ can be calculated using linear interpolation along the span.

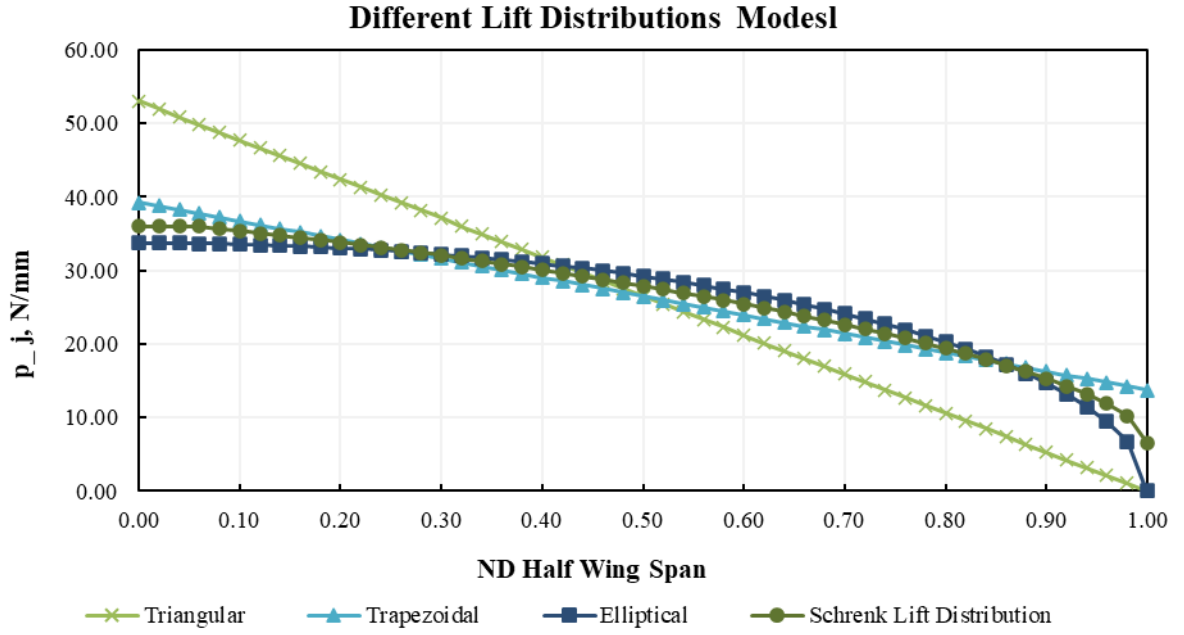


Figure 29 Different types of distribution models applied for the spanwise lift distribution in sample Flight Case-421 for the configuration presented in Figure 28.

4.2.2 Trapezoidal Distribution

The load per unit length (N/mm) at the root of the wing, $P_{Trap,root}$, a simple trapezoidal load distribution shape, as depicted in Figure 29, can be formulated as in Eq.(4-2).

$$P_{Trap,root} = (2 \times F) / (b \times (1 + \tau_{Trap})) \quad (4-2)$$

Where the ratio (τ_{Trap}) between the root and the tip values of $P_{Trap,j}$ is $\tau_{Trap} = P_{Trap,Tip} / P_{Trap,Root}$. When the trapezoidal distribution is preferred, the values of τ_{Trap} are set to 0.35, 0.1, and 0.1 in the tool for wing lift distribution, fuselage mass distribution and wing mass distribution, respectively. $P_{Trap,j}$ can then be calculated using linear interpolation along the wing between $P_{Trap,Tip}$ and $P_{Trap,Root}$.

4.2.3 Elliptical Distribution

The load per unit length (N/mm) at the root of the wing for elliptic distribution ($P_{Ellip,Root}$), as shown in Figure 29, can be computed by Eq.(4-3). Then the value of $P_{Ellip,j}$ along the wing can be calculated with Eq.(4-4).

$$P_{Ellip,Root} = (4 \times F)/(b \times \pi) \quad (4-3)$$

$$P_{Ellip,j} = P_{Ellip,Root} \times \sqrt{(1 - A_j^2)} \quad (4-4)$$

4.2.4 Schrenk's Lift Distribution Method

In addition to the three distribution models, Schrenk's lift distribution model is also utilized in the tool. Schrenk's lift distribution method details can be found in (Howe, 2004). If the original wing planform is converted into an equivalent elliptic planform, the chord variation along the wing can be calculated with Eq.(4-5).

$$c_{Ellip,j} = \frac{4 \times S_{LS}}{\pi \times b} \times \sqrt{1 - \left(\frac{2 \times y_{LS,j}}{b}\right)^2} \quad (4-5)$$

Schrenk's chord distribution is the average of the original and the elliptic chords and is given in Eq.(4-6).

$$c_{Schrenk,j} = (c_{Ellip,j} + c_j)/2 \quad (4-6)$$

Once $c_{Schrenk,j}$ is calculated, then the total lift can be redistributed along the LS proportional to the value of $c_{Schrenk,j}$ as given in (4-7).

$$P_{Schrenk,j} = (L_{LS,Case} / (S_{LS})) \times c_{Schrenk,j} \quad (4-7)$$

The graph in Figure 29 presents the four types of lift distributions explained in this section for sample flight Case-421.

4.2.5 Low-Order CFD Models

If higher fidelity levels are aimed, external low-order aerodynamic models such as Vortex lattice or panel methods can be integrated into the tool to generate more realistic and accurate lift distributions in the spanwise and chordwise direction. The geometric models of the lifting surfaces presented in this work are compatible with these aerodynamic tools due to their discrete design with sections of the lifting surfaces.

4.2.6 Strip Load Calculation

Finally, distributed loads can be assumed as strip loads and positioned at sections of the components for simplicity in the spreadsheet calculations, as shown in Figure 30. F_{Strip} at j th section then can be formulated with Eq. (4-8)

$$F_{Strip,j} = (b_{Bay,j} * (P_j + P_{j+1})) / 2 \quad (4-8)$$

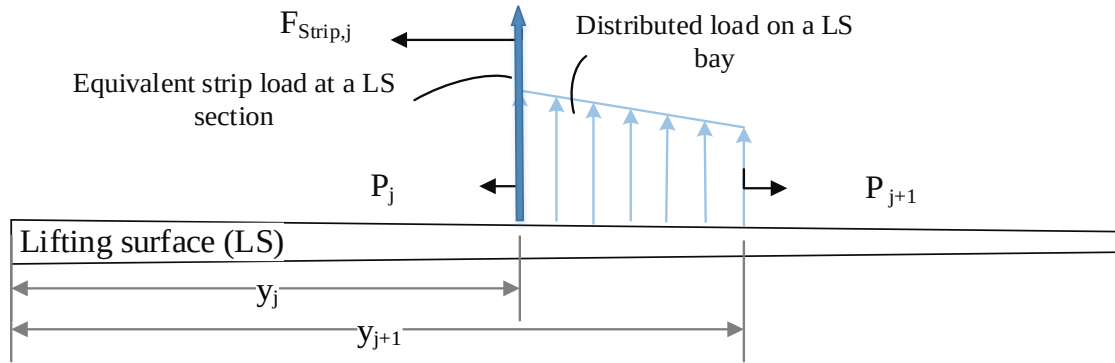


Figure 30 Strip load calculation of distributed loads for spreadsheet methods.

4.3 Aircraft Mass Moment of Inertia Model

All aircraft's mass moment of inertia is required for rolling case calculations. Hence, a new MMol model was generated to be computationally efficient and have sufficient accuracy. It is assumed that the aircraft components can be broken down into simplified geometric shapes, as shown in Figure 31, and their MMol can be calculated easily at their local CG and then carried to any reference line with the parallel axes theorem. In the idealized model in Figure 31, all the sections are evolved into cylinders or cuboids. The MMol of the entire aircraft around x , y and, z axes can be obtained by summing the MMol of all main components as explained below.

The fuselage is divided into 50 simplified cylinders as default for geometric simplification. Each cylinder has an equal diameter with the relevant section of the fuselage. The mass of the cylindrical sections is usually unknown at the initial design stage; however, if the total mass of the fuselage is estimated or known, it can then be distributed in an assumed shape, as explained above.

Wings and fuel tanks are simplified into cuboids, as shown in Figure 31. The dimension of the simplified cuboid wing section shape is calculated from the thickness and the chord of the actual wing, while that of the fuel tank is calculated from the thickness and the

width of the relevant wing box section. The mass of the wing can either be calculated with a proportional, empirical, or simple PB approach to initialize the tool, and it can be distributed with the simple methods discussed above. For a more accurate MMol, an iterative approach can be used, and the distribution shape can be obtained from the outcome of the detailed MEs.

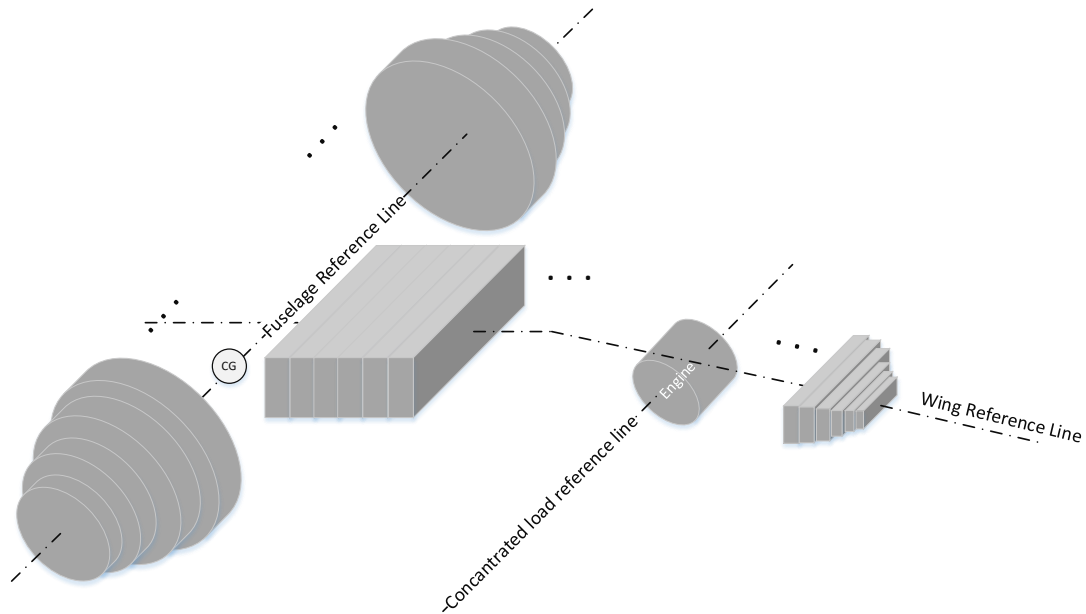


Figure 31 Simplified geometric model for mass moment of inertia calculation.

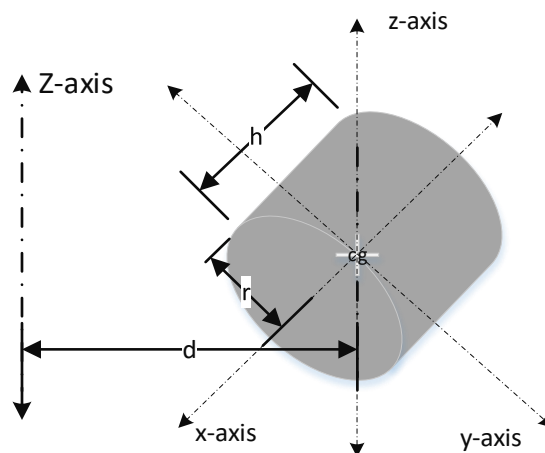


Figure 32 A sample illustration for parallel axes theorem.

The parallel axes theorem is used to calculate the MMol of a body about any reference axis parallel to any of the local axes of the body. If, for example, the MMol of the cylinder

in Figure 32 is calculated about the z-axis, its MMol around the Z-axis can then be calculated using the theorem with Eq. (4-9).

$$I_{ZZ} = I_{zz} + Md^2 \quad (4-9)$$

in which M is the mass of the body, I_{zz} is the MMol of the body around the axis-z, which passes through the CG of the body, and d is the distance between the z-axis and Z-axis.

4.3.1 MMol of the Fuselage

The local MMol of each of the cylindrical sections in the simplified fuselage model can be calculated using Eq. (4-10) and (4-11), and they can then be carried to the CG of the aircraft using parallel axes theorem with Eq. (4-9).

$$I_{xx} = \frac{1}{2} * M * r^2 \quad (4-10)$$

$$I_{yy} = I_{zz} = \frac{1}{12} * M * (3r^2 + h^2) \quad (4-11)$$

in which r , h and M are the radius, height, and mass of the cylinder, respectively. The dimensions r and h are illustrated in Figure 32.

The mass of each cylindrical section can be estimated using mass distribution models if a more realistic distribution shape is not available. The sample trapezoidal mass distribution of the fuselage is illustrated in Figure 33. The total mass of the fuselage can be estimated using existing empirical methods, as some samples are listed in Table 13.

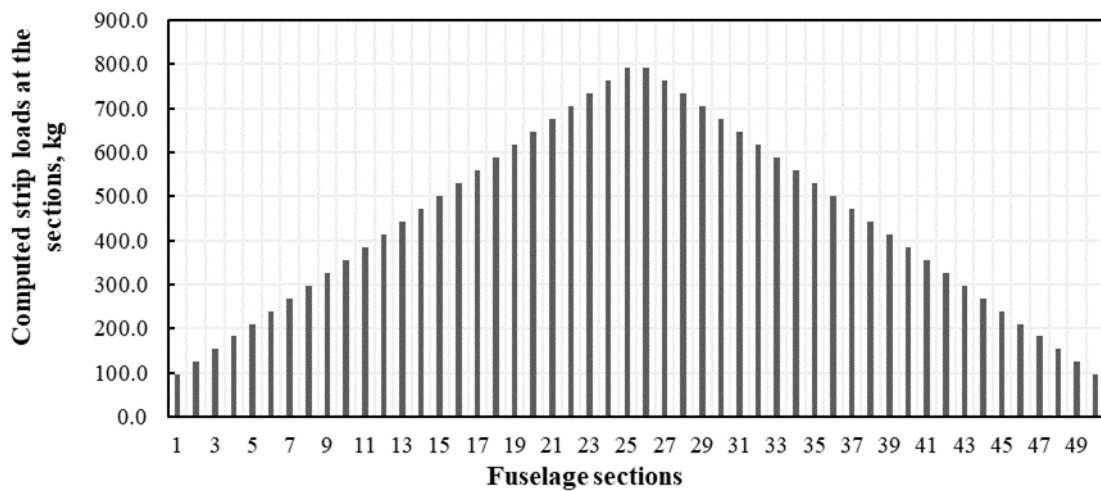


Figure 33 Trapezoidal mass distribution of the total fuselage mass.

4.3.2 MMol of the Wing and Fuel

MMol of a simplified cuboid shape of the wing or the fuel tanks about the x, y and z axes can be calculated with Eqs. (4-12) - (4-14). The dimensions of a , b and c of the cuboid are given in Figure 34. a is the wing section length for both the wing and the fuel tank calculations. b is the maximum thickness of the wing or the wing box, respectively at the local sections. Finally, c is the width of the local wing box section for the fuel tank calculations.

$$I_{xx} = \frac{1}{12} * M * (a^2 + b^2) \quad (4-12)$$

$$I_{yy} = \frac{1}{12} * M * (b^2 + c^2) \quad (4-13)$$

$$I_{zz} = \frac{1}{12} * M * (a^2 + c^2) \quad (4-14)$$

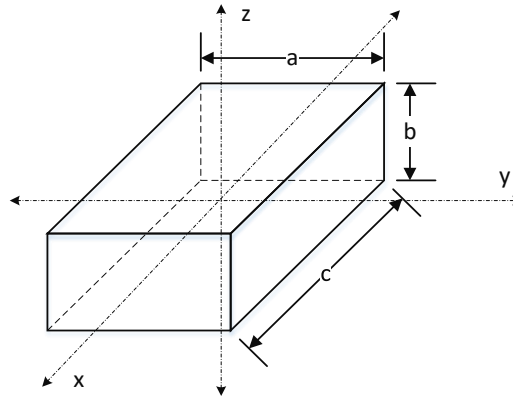


Figure 34 Illustration for MMol of cuboid.

At the initial design stage of an aircraft, the wing's mass can be calculated with the proportional approach, empirical equations, or physics-based approaches, as listed in Table 13. If the total mass of the wing is computed with one of these methods, the wing mass can be assumed to be distributed in a trapezoidal shape, as illustrated in Figure 35. This distribution can be calculated with a similar approach in Eqs. (4-2).

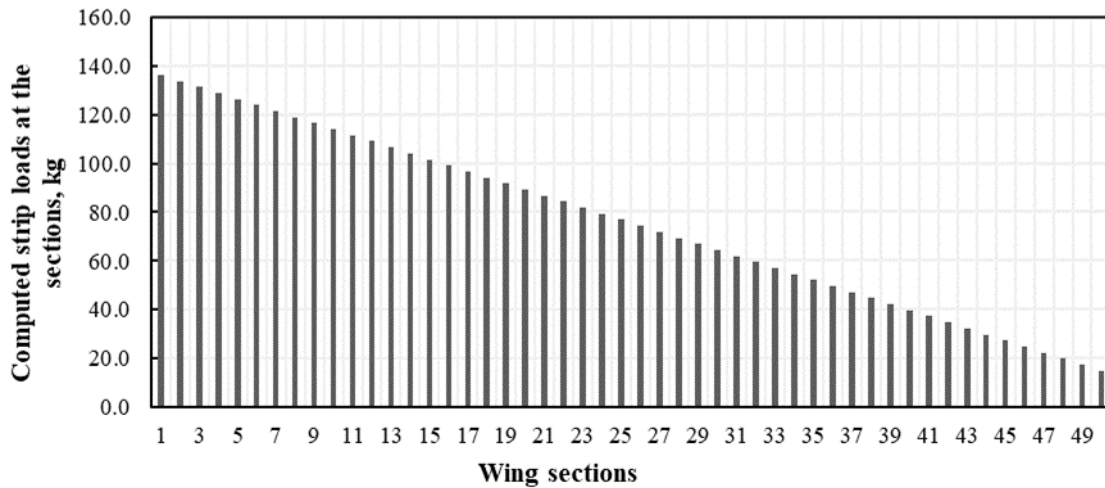


Figure 35 Wing simplified mass distribution for MMol.

It should be noted that after running the new ME tool, the generated mass distribution for the wing mass can then be used in the MMol analysis for a more accurate MMol calculation.

The fuel mass distribution is calculated with a simplified approach in Section 4.4.2.1 Fuel; this load distribution will be used in MMol calculation, as shown in Figure 36.

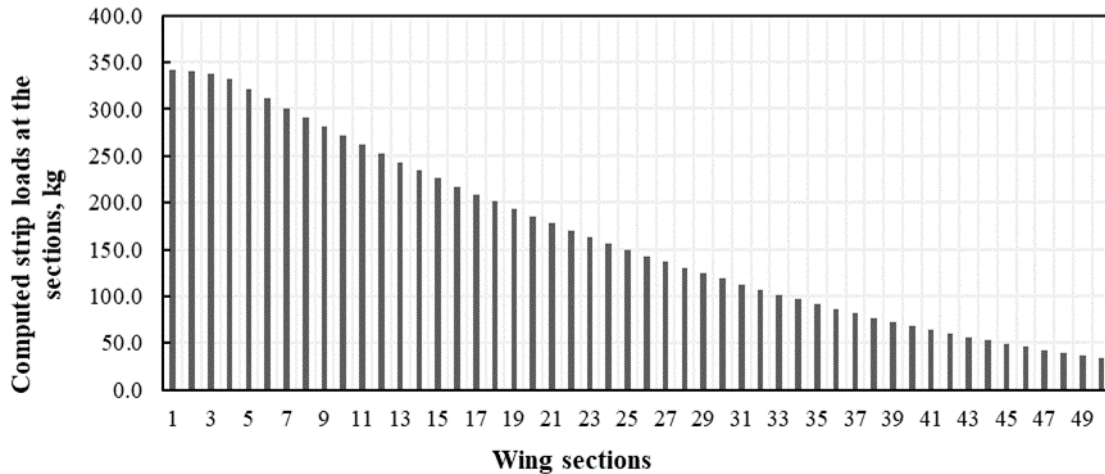


Figure 36 Fuel mass distribution for MMol.

4.3.3 MMol of Concentrated Load

All the concentrated loads, such as engines, landing gears, and drop tanks, are assumed to have a cylindrical shape, as shown in Figure 31. Their MMol in their CG can be calculated with (4-10) and (4-11). The local MMol results can then be carried to a reference line using the parallel axes theorem in Eq.(4-9).

Finally, the MMol of the aircraft around x, y and z axes can be obtained by summing the MMol of all main components after calculating them with the methods presented.

4.4 Inertia Loads

Inertia loads can be grouped as concentrated and distributed loads. The computations of those loads are explained in this section.

4.4.1 Concentrated Masses

In a transport aircraft, the main concentrated loads are generally caused by the mass of the engines, landing gear, and fuel drop tanks. In the developed model, an unlimited number of concentrated loads can be defined along the wing and positioned in the 3D axes system. Concentrated masses are either known or can be calculated by the simple empirical method at the initial design stage. They cause inertia loads during the symmetric, rolling, and combined manoeuvre cases. The total concentrated weight at j th wing section can be computed with Eq.(4-15)

$$W_{Tot\ Con,j} = W_{Engine,j} + W_{MLG,j} + W_{DTanks,j} \quad (4-15)$$

4.4.2 Distributed Masses

Distributed inertia loads in this study are grouped as fuel and wing weight. The wing weight covers any weight in the wing group except the distributed fuel and concentrated loads. Distributed ILs are mainly caused by the masses of fuel stored inside the wing, the wing's structure, fuel piping systems, hydraulic and electric systems, painting, secondary components, and other miscellaneous items.

4.4.2.1 Fuel Mass

The distributed fuel load in each wing bay can be rearranged as concentrated strip loads for simplicity and located at wing sections, as shown in Figure 30. A simplified internal-wing fuel tank geometric model is created to compute the magnitude of the strip load caused by the fuel mass, as illustrated in Figure 37.

The geometric shape in Figure 37 is a truncated irregular hexagonal pyramid, and its volume can be calculated with Eq.(4-16), which also gives the maximum volume of this geometric space for fuel storage, $V_{Fuel\ tank,Max,j}$.

$$V_{Fuel\ tank,Max,j} = \frac{1}{3} * b_{Bay,j} \quad (4-16)$$

$$* \left(A_{6-sided,j} + A_{6-sided,j+1} + \sqrt{A_{6-sided,j} * A_{6-sided,j+1}} \right)$$

In Eq. (4-16), $A_{6-sided,j}$ and $A_{6-sided,j+1}$ are the bottom and top surface areas of the pyramid, and their formulations are presented in Eq. Eq.(4-17).

$$A_{6-sided,j} = w_{wb,j} * t_{wb,j} - w_{wb,j} * \frac{t_{wb,j} - t_{wb,edge,j}}{2} \quad (4-17)$$

$$A_{6-sided,j+1} = w_{wb,j+1} * t_{wb,j+1} - w_{wb,j+1} * (t_{wb,j+1} - t_{wb,edge,j+1})/2$$

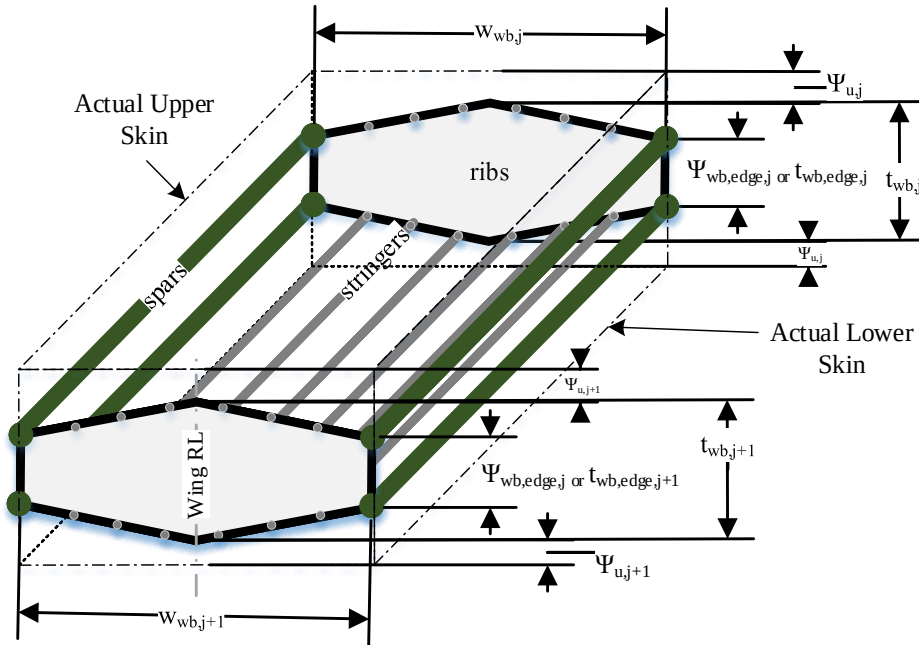


Figure 37 Simplified wing fuel tank design.

$V_{Fuel\ tank,Max,j}$ is reduced due to the fuel tank structure and any other device in the internal space of the wing box. Therefore, a fuel tank efficiency factor of $\eta_{tank,j}$ is introduced to account for this volume reduction. Using Eq. (4-18), the capacity of the fuel tanks in each wing section, $V_{Fuel\ tank,j}$, can be adjusted with $\eta_{tank,j}$ by the designer or optimization algorithms. After calculating the $V_{Fuel\ tank,j}$, mass and weight of the fuel, $M_{Wing\ fuel,Casej}$, $W_{Wing\ fuel,Casej}$ can be computed by Eq.(4-19) and Eq.(4-20), respectively.

$$V_{Fuel\ tank,j} = V_{Fuel\ tank,Max,j} * \eta_{Tank} \quad (4-18)$$

$$M_{Wing\ fuel,Casej} = V_{Fuel\ tank,j} * \rho_{Fuel} * Case_{Fuel\ percent} \quad (4-19)$$

$$W_{Wing\ fuel,Case,j} = M_{Wing\ fuel,Case,j} * g \quad (4-20)$$

where $Case_{Fuel\ percent}$ accounts for the percentage of the fuel remaining in the wing fuel tank as $M_{Wing\ fuel,Case,j}$ and $W_{Wing\ fuel,Case,j}$ are changed in each flight case.

4.4.2.2 Wing Mass

Another source of the distributed inertia loads is the wing group mass. The wing group mass (WGM) should be estimated to initialise the wing tool, as the wing box mass is a dependent value of the wing group mass. One of the options presented in Table 13 can be chosen for initialisation. It will be shown in the next chapters that even if the first wing group ME is not accurate, the ME tool converged to a final value less than 3 iteration steps, or even the direct solver provides sufficient accuracy.

Table 13 Wing weight initialising options.

Initial WGM Methods
0- TOGW Fraction Method
1-P. York,1980 (York and Labell, 1980)
2-Chizotto,2017 (Chiozzotto, 2017)
3-D. Howe-Simple (Howe, 2000)
4-Raymer-General (Raymer, 2018)
5-Raymer-Transport (Raymer, 2018)
6-Jenkinson (Jenkinson, Simpkin and Rhodes, 1999)
7-Cranfield Emp.
8-Actual Mass
9-User Input

In Table 13, the TOGW fraction method is the simplest initialisation approach. Higher fidelity empirical methods can also be used to initialise the tool. Users can also introduce the initial wing group mass.

After the first WGM estimation, the distribution shape of the mass is still unknown. Hence for the initialising or the non-iterative solver, it is assumed that the wing mass is distributed in a trapezoidal shape along the wing.

The distributed load per unit length at the root is $P_{Wing,Trap,Root}$ while that of the load at the tip is $P_{Wing,Trap,Tip}$. Those values can be calculated with the help of Eqs. (4-2) where $W_{Wing,Initial,Chosen}$ is the first wing weight estimation to be distributed and can be calculated with one of the methods shown in Table 13. τ_{Trap} is the ratio of the $P_{Wing,Trap,Tip}$ to $P_{Wing,Trap,Root}$. By default, τ_{Trap} value is 0.1; however, it can be changed for a more representative distribution. The strip load in j th bay of the wing caused by the weight distribution, $W_{Wing,Trap,j}$ can be calculated with Eq.(4-21).

$$W_{Wing,Trap,j} = (P_{Wing,Trap,j} + P_{Wing,Trap,j+1}) * b_{Bay,j} / 2 \quad (4-21)$$

4.4.3 Total Inertia Loads

The total weight $W_{Total\ Inertia,j}$ caused by the distributed and concentrated weights is given in Eq.(4-22).

$$W_{Total\ Inertia,j} = W_{Wing,Trap,j} + W_{Wing\ fuel,Case,j} + W_{Tot\ Con,j} \quad (4-22)$$

The inertia loads (ILs) at j th wing section during the symmetric, $F_{Man,j}$, and the rolling, $F_{Roll,j}$, manoeuvre cases can be computed with Eq. (4-23) and Eq.(4-24)

$$F_{Man,j} = W_{Total\ Inertia,j} * n_{Sym\ Loading} \quad (4-23)$$

$$F_{Roll,j} = \frac{W_{Total\ Inertia,j}}{g} * \frac{y_{Section,j} + y_{Section,j+1}}{2} * \bar{p} \quad (4-24)$$

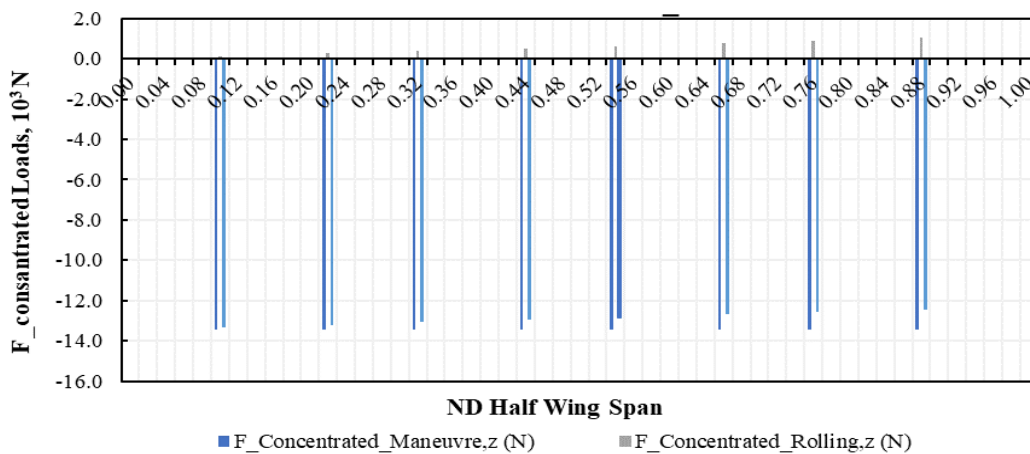


Figure 38 The graph indicates the loads generated by concentrated loads in the symmetric and rolling manoeuvres and their combination during the sample Flight Case-421 for the configuration presented in Figure 28.

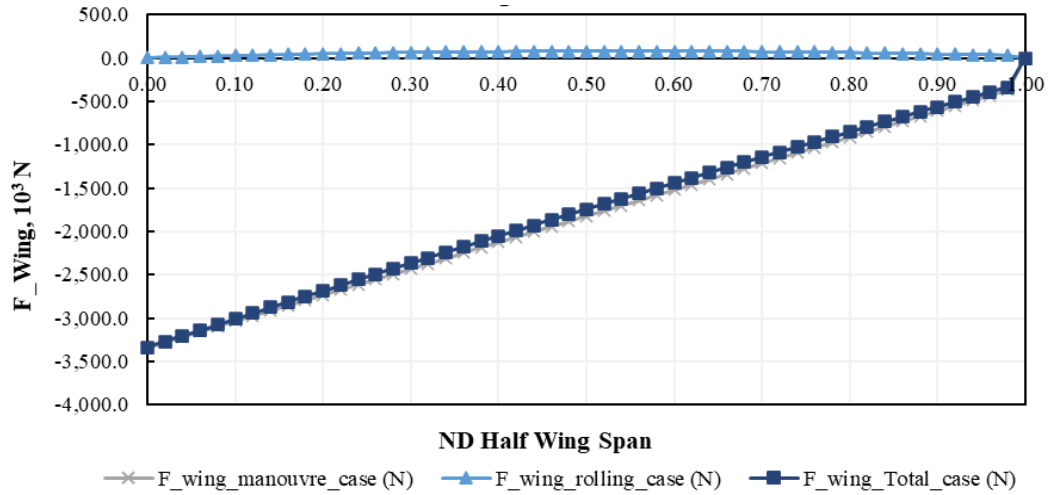


Figure 39 The graph indicates the loads generated by wing inertia in the symmetric and rolling manoeuvres and their combination during the sample Flight Case-421 for the configuration presented in Figure 28.

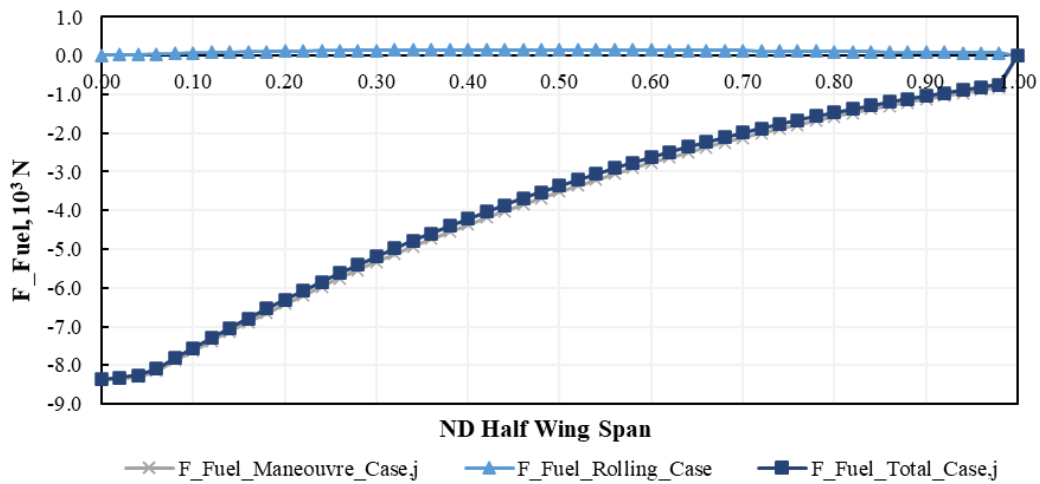


Figure 40 The graph shows the loads generated by fuel mass in the symmetric and rolling manoeuvres and their combination during the sample Flight Case-421 for the configuration presented in Figure 28.

Eq. (4-24) assumes that the CG of each concentrated load or a strip load on each wing bay is in the middle of the wing bay. Finally, the total IL due to manoeuvres can be calculated with Eq. (4-25) for each loading case.

For just symmetric manoeuvre case,

$$F_{Total\ inertia,j} = F_{Man,j} \quad (4-25)$$

For combined case,

$$F_{Total\ inertia,j} = \frac{2}{3} * F_{Man,j} + F_{Roll,j}$$

Utilising Eqs. (4-16)-(4-25), inertia loads can be calculated as indicated in Figure 38 for concentrated loads, Figure 39 for wing mass, and Figure 40 for wing distributed fuel mass for a sample flight case.

4.5 Aerodynamic Loads on the Wing

This section will present the computation procedure of the aerodynamic loads emerging in symmetric and combined flight cases.

4.5.1 Spanwise Lift Distribution

Lift distribution along a 3D wing is a complex phenomenon. An accurate prediction of the distribution requires high-fidelity fluid models, which tend to cause long computational time. In the initial design stage, simple lift distribution models may provide sufficiently accurate results. Hence, simple models of triangular, trapezoidal, elliptical and Schrenk's lift distributions are introduced, and low-order CFD models are discussed in Section - 4.2 Load Distribution Models.

To calculate the total generated lift in each flight case, It was assumed that the wing generates a total lift of $L_{Wing,Case}$ whose magnitude is equal to the inertia force of $F_{Aircraft,Case}$ acting on the aircraft, but in the opposite direction, as given in Eq.(4-26). After computing $-L_{Wing,Case}$, it can be distributed with the simple models presented previously.

$$-L_{Wing,Case} = F_{Aircraft,Case} = W_{Aircraft,Case} * n_{Loading} \quad (4-26)$$

4.5.2 Loads in Rolling

In this section, the methods and assumptions for the calculation of aileron load and aerodynamic damping will be presented.

The aircraft's mass moment of inertia (MMol) should be known to calculate the rolling load. As the value of MMol changes with the evolution of the design, a low-fidelity, computationally efficient MMol inertia model is developed and presented in Section - 4.3 (Aircraft Mass Moment of Inertia Model).

4.5.2.1 Aileron Loads

The aerodynamic loads generated by the ailerons are calculated following the requirements in CS-25 (EASA, 2020). Ailerons should generate enough moments, $\overline{L}_\xi$, around the longitudinal axis of an aircraft to achieve rolling by generating additional lift, $\bar{Z}_\xi$. The required aileron lift, $\bar{Z}_\xi$, can be found by the general equation in Eq. Eq.(4-27).

$$\bar{Z}_\xi = \frac{\overline{L}_\xi}{2y_\xi} \quad (4-27)$$

In which, y_ξ is the equivalent moment arm of a high-speed (HS) and a low-speed (LS) aileron and $\overline{L}_\xi$ is the required rolling moment and given in Eq.(4-28).

$$\overline{L}_\xi = 0.5\rho_{Air}V_{TAS,Case}^2 S_{Aileron} b \xi L_\xi \quad (4-28)$$

The shape of the spanwise lift distribution of an aileron has a minimal effect on the structural design of the wing; hence the aileron lift distribution is assumed to have a rectangular shape, as shown in Figure 41.

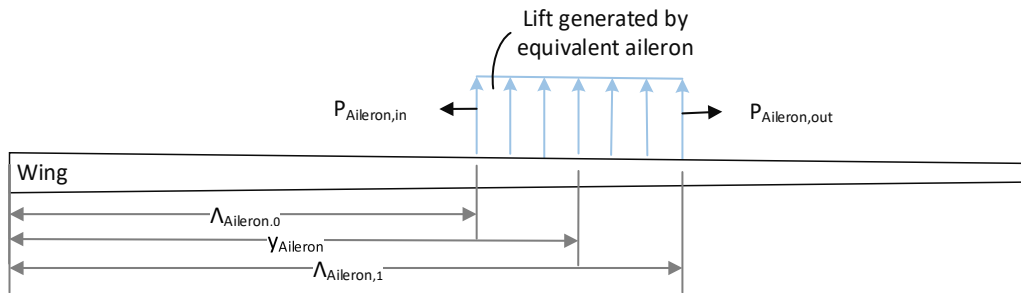


Figure 41 High (HS) and low speed (LS) aileron positions along the wingspan and their lift distributions.

At the initial design stage, there are many unknown parameters, and optimization of the positions of HS and LS ailerons with the objectives of aerodynamic performance or weight reduction can cause complexities. Hence, it was assumed that the additional lift generated by those ailerons could be represented with one equivalent aileron, as shown in Figure 41. If the inboard and outboard locations of the equivalent aileron are known, y_ξ can be computed with Eq. (4-29).

$$y_\xi = b \left(\frac{\lambda_{Aileron,0} + \lambda_{Aileron,1}}{4} \right) \quad (4-29)$$

Additional total lift or the force slope $\bar{Z}_\xi$ can be computed with Eq.(4-30). If it is assumed to have a rectangular distribution, the lift per unit length $P_{Ail,j}$ can be calculated by Eq.(4-31),

$$\bar{Z}_\xi = \frac{(\rho_{Air} V_{TAS,Case}^2 S_{Aileron} \xi L_\xi)}{(\lambda_{Aileron,0} + \lambda_{Aileron,1})} \quad (4-30)$$

$$P_{Ail,j} = \frac{\bar{Z}_\xi * b}{2 * (\Lambda_{Ail,out} - \Lambda_{Ail,in})} \quad (4-31)$$

The strip load applied to the wing section due to the aileron lift can be found using Eq.(4-32)

$$F_{Ail,j} = b_{Bay,j} * P_{Ail,j} \quad (4-32)$$

4.5.2.2 Aerodynamic Damping

Due to the rolling, each wing section has a local velocity ($p_{Case} * y_{Section}$) which is perpendicular to the flight path, and the local effective angle of attack of those sections is changed by $\tan^{-1}(p_{Case} * y_{Wing,j} / V_{Case})$ (Howe, 2004). Hence, the local lift coefficient, $C_{L,p,j}$, can be written as,

$$C_{L,p,j} = \frac{(p_{Case} * y_{Wing,j})}{(V_{Case})} a_{1,j} \quad (4-33)$$

in which $a_{1,j}$ is the lift curve slope of the wing bay. If the equation is updated by Schrenk's hypothesis (Howe, 2004) and then multiplied by $0.5\rho V^2$, the spanwise aerodynamic damping load can be written as in Eq.(4-34). And the strip load at j th wing bay due to the distributed damping load can be calculated with Eq.(4-35).

$$P_{Damping,j} = 0.5\rho_{Air} V_{Case} p_{Case} y_{Wing,j} a_{1,j} \left(0.5c_j + \frac{2S_{Wing}}{\pi b} \sqrt{1 - (2y_{Wing,j}/b)^2} \right) \quad (4-34)$$

$$F_{Damping,j} = (b_{Bay,j} * (P_{Damping,j} + P_{Damping,j+1})) / 2 \quad (4-35)$$

Figure 42 shows the three types of distributed aerodynamic loads, including wing lift, aileron lift and aerodynamic damping from the sample flight Case-437.

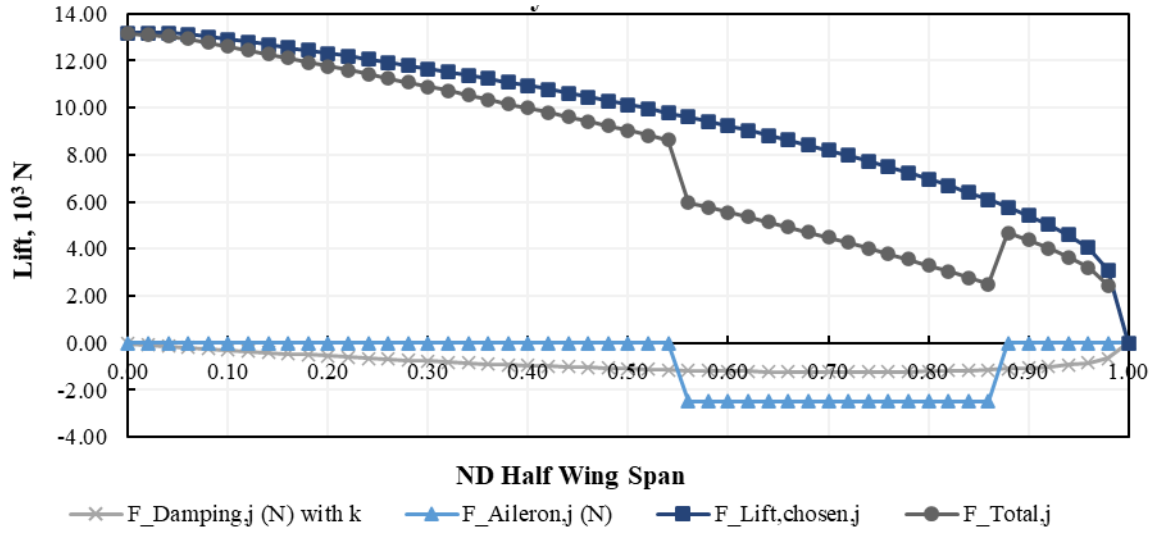


Figure 42 The graph presents aerodynamic load distribution on a wing, including lift distribution, the lift generated by an equivalent aileron and aerodynamic damping distribution for sample Flight Case-437 for the configuration presented in Figure 28.

4.5.3 Chordwise Lift Distribution

For this study, a wing bay moment coefficient, C_m , is used to find the torsion caused by the lift distribution along the wing's chord. For better accuracy, the value of C_m can be obtained from low-order CFD tools. This value can be the default if a fast ME calculation is required. The value of C_m should be chosen carefully, as the wing weight is sensitive to this parameter.

4.5.4 Total Loads

The distributed lift loads can be assumed as strip loads, as explained before and $F_{Lift,j}$ can be calculated using Eq.(4-8) for any lift distribution model selected. The total aerodynamic load at the j th section of the LS can be computed with Eq.(4-36).

$$F_{Aero,j} = F_{Damping,j} + F_{Ail,j} + F_{Lift,j} \quad (4-36)$$

4.6 Aerodynamic Loads on Truss Components

In a truss braced wing (TBW) aircraft, although the truss components (jury, offset and strut) are structural elements and are not designed to generate lift, they can still be prone to the aerodynamic loads due to the change in the direction of airflow as a result of the gust or the manoeuvre of the aircraft.

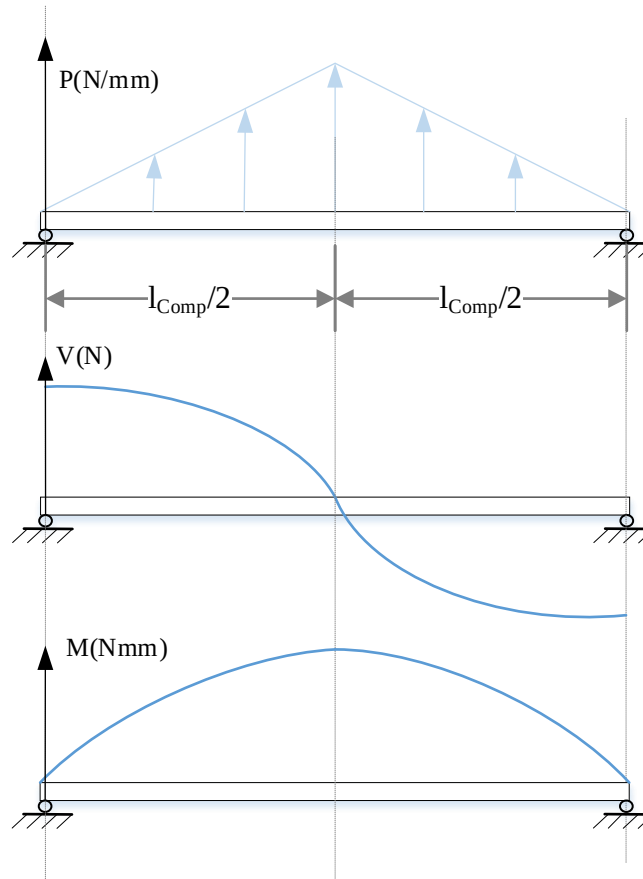


Figure 43 The truss components' Lift, shear, and moment distribution models.

In this study, truss components are assumed to have a symmetrical airfoil shape and be positioned with zero incidence angle to the relative wind. A conservative approach is considered to calculate the aerodynamic load acting on these components. The total lift generated by these components is computed using Eq.(4-37).

$$L_{Comp.} = C_{L,max,Comp.} q_{\infty} S_{Comp.} \quad (4-37)$$

where $C_{L,max,Comp.}$ is the maximum lift coefficient of the finite component, q_{∞} is dynamic pressure of the flight case and $S_{Comp.}$ is the planform area of the component. As a default value $C_{L,max,Comp.}$ is taken 1.2 in the tool. After finding the total lift generated by the

component, it was assumed that the lift is distributed in a triangular shape, as shown in Figure 43. The triangular shape is chosen to generate the highest possible bending moment at the centre of the component, as illustrated in the shear and moment diagram in Figure 43, and the load-carrying box of the component is sized to resist this bending moment. The total bending moment at the middle of the component can be computed using Eq.(4-38).

$$M_{Comp.} = \frac{L_{Comp.}l_{Comp}}{6} \quad (4-38)$$

The following chapter will explain the sizing of the offset and jury to resist this bending moment using a rectangular load-carrying box model.

4.7 Strut Loads

The reaction forces caused by the strut are classified as a concentrated load in this study. When this reaction force is calculated, it is applied to the wing at the strut wing joint using the procedure explained in this chapter. The strut reaction forces' calculation method is presented in Section 5.3 (Solutions to the Indeterminate SBW Problem).

4.8 Shear and Moment Calculations

After computing all the loads caused by the inertia and aerodynamics, shear and moment about three axes across the wing can be found. As a result of this, simple numerical spreadsheet approaches used for these calculations will be explained.

4.8.1 Total Shear Load

The total shear load ($V_{Total,j}$) at each wing section can be obtained with Eq. Eq.(4-40), where $F_{Total,j}$ is given in (4-39).

$$F_{Total,j} = F_{Total\ Inertia,j} + F_{Aero,j} \quad (4-39)$$

$$V_{Total,j} = F_{Total,j} + V_{Total,j+1} \quad (4-40)$$

The shear loads generated by different loads are plotted and shown in Figure 44 for sample Flight Case-421. If all the shear load values at the wing sections are summed, then $V_{Total,z,j}$ can be calculated and plotted as illustrated in Figure 45.

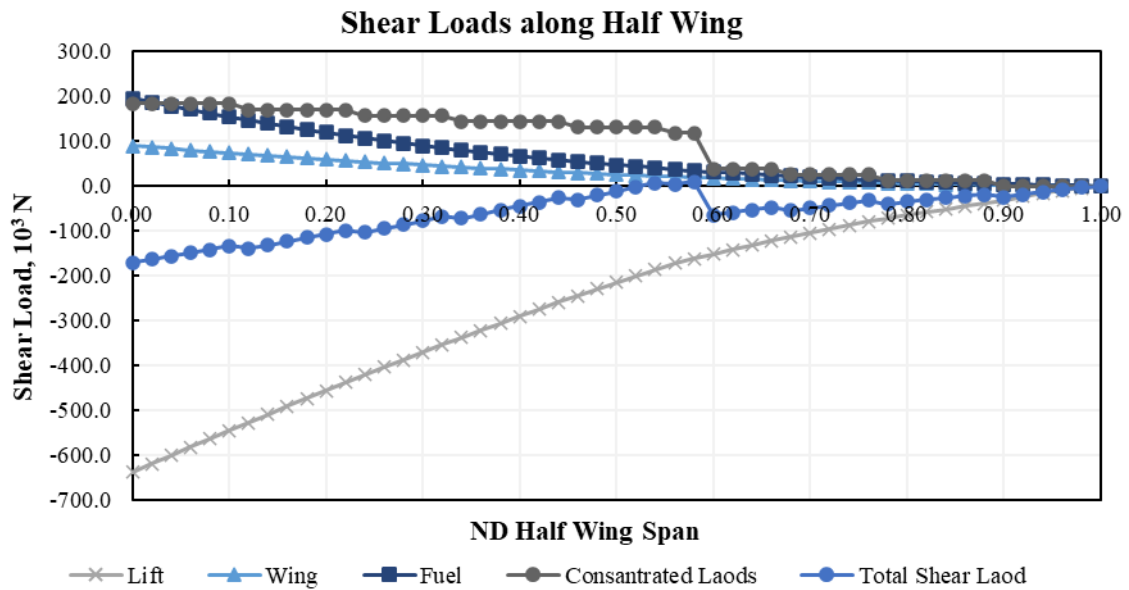


Figure 44 Shear load generated by all types of distributed and the concentrated load along the wing is presented in the graph in sample Flight Case-421 for the configuration presented in Figure 28.

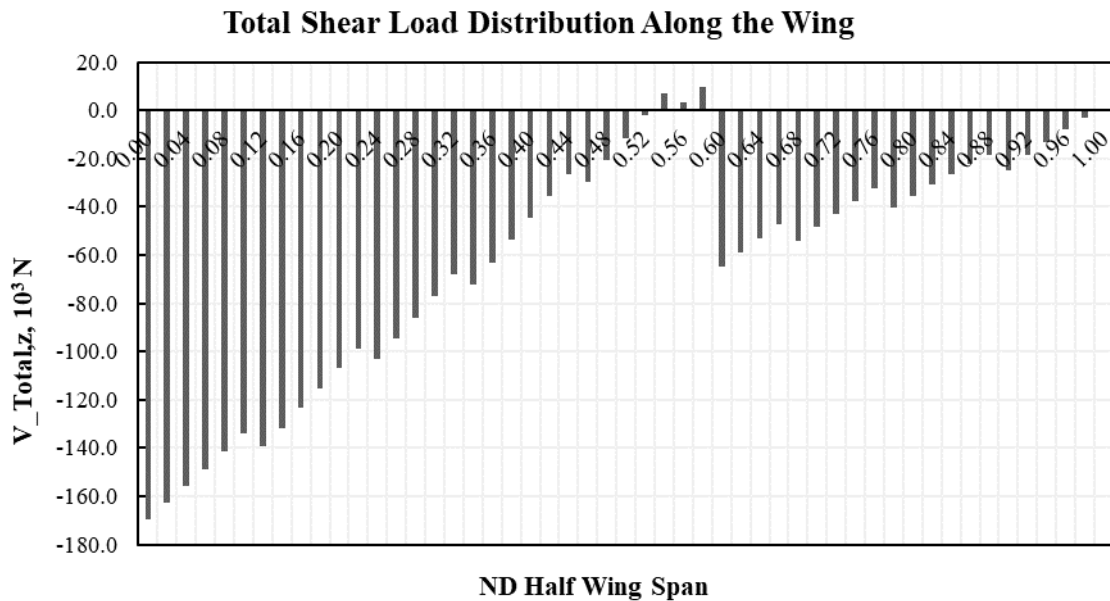


Figure 45 Total shear load distribution along the wing in sample Flight Case-421 for the configuration presented in Figure 28.

4.8.2 Total Bending Moment

If the shear load variation between the wing sections is assumed to be linear, then the trapezoidal area under the shear graph of a wing bay can be used to compute the moment increment between the wing sections due to shear load as given in Eq.(4-41).

$$\Delta M_{Total,j} = (V_{Total,j} + V_{Total,j+1}) * b_{Bay,j}/2 \quad (4-41)$$

After finding the total moment increment in a wing bay, $\Delta M_{Total,j}$, the total bending moment along a straight (without angle change in reference line) lifting surface (wing), $M_{Total,j}$, can be formulated as (4-42). The effect of load path change on the moment calculations will be explained in Section 4.8.4 (Vector Transformations).

$$M_{Total,j} = \Delta M_{Total,j} + M_{Total,j+1} \quad (4-42)$$

Eq.(4-41) and (4-42) are general formulation for calculating the bending moment around the x and z axes of the lifting surface (wing), and it is suitable for spreadsheet applications. A sample total bending moment graph is illustrated in Figure 46 in sample Flight Case-421 for the configuration presented in Figure 28.

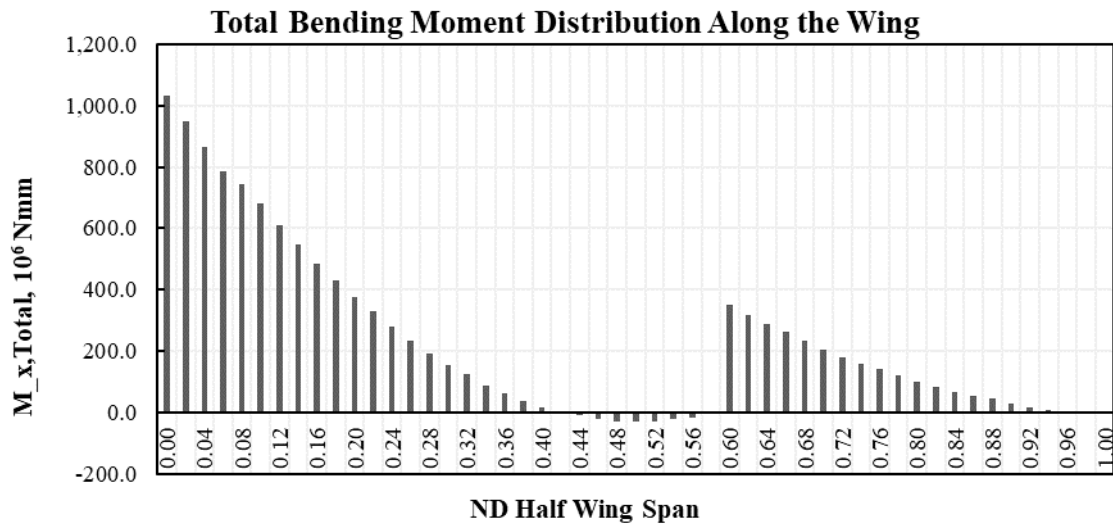


Figure 46 Total bending moment variation across the wing around the x-axis in sample Flight Case-421 for the configuration presented in Figure 28.

4.8.3 Torsion Calculation

It is assumed that the CG of the wing and the fuel masses coincide with the shear centre of the wing box; hence they do not cause torsion. Torsion can be caused by the engine,

landing gears, drop tanks, engine thrust and distributed loads, including aerodynamic and inertia which are not located at the shear centre of the wing (lifting surface).

The pressure centre of the lift does not overlap with the shear centre of the wing box; hence it causes torsion. The torsion increment caused by the pitching moment of the lifting surface (LS)($\Delta M_{y,Cm,j}$) bay acting at the wing bay quarter chord can be calculated with Eq.(4-43).

$$\Delta M_{y,Cm,j} = (MAC_{Bay,j} * S_{Bay,j} * C_{m,Bay,j} * q) \quad (4-43)$$

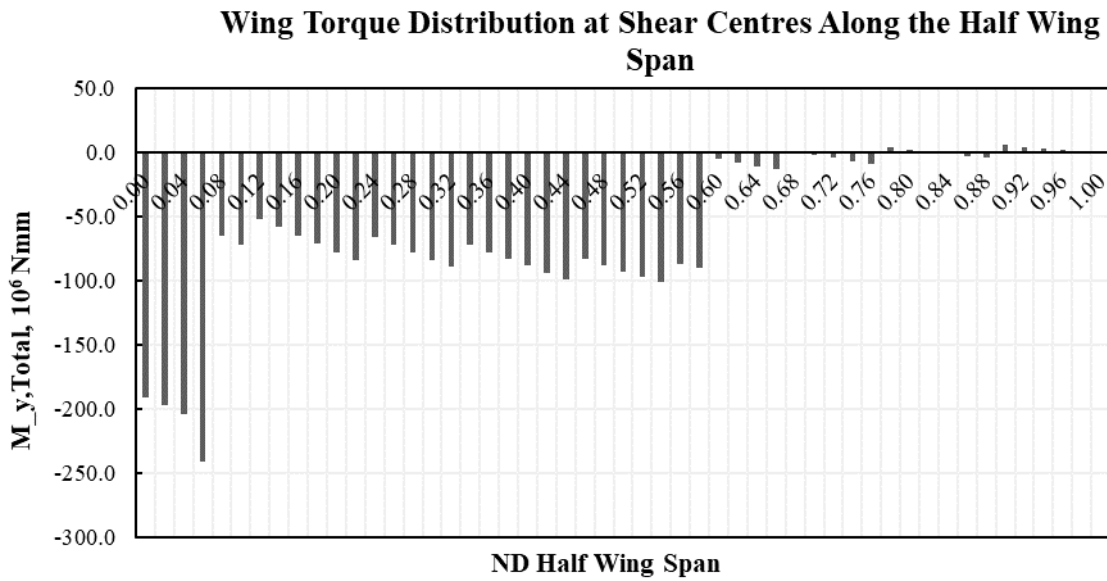


Figure 47 Total torsion variation along the wing in sample Flight Case-421 for the configuration presented in Figure 28.

Equation (4-43) assumes that the lift generated by the LS (wing) bay is moved from the pressure centre of the LS bay to the quarter chord of it with a torque generated by the lift. The lift and the torque at the quarter chord should be later moved to the shear centre of the LS box. When repositioned, the lift will generate a new torque at the shear centre. The computation of the additional torque between the quarter chord and the LS centre is taken into account with $F_{Aero,j} * d_{T2,Arm,j}$ in Eq.(4-44). It can be seen in Eq.(4-45) that two torque products are summed to find the final torque caused by the lift at the shear centre. This procedure is followed to make the model more compatible with aerodynamic tools, which usually provide the moment coefficient at the quarter chord of the LS.

Finally, the torsion increment at the bay of an LS ($\Delta M_{y,Tot,j}$) is given in Eq.(4-44) and total torsion at j th section of the LS ($M_{y,Tot,j}$) can be computed with Eq.(4-45). Figure 47 illustrates a sample torsion graph in sample Flight Case-421 for the configuration presented in Figure 28.

$$\Delta M_{y,Tot,j} = F_{Total\ Inertia,j} * d_{T1,Arm,j} + F_{Aero,j} * d_{T2,Arm,j} + F_{Strut,j} * d_{T3,Arm,j} \quad (4-44)$$

$$M_{y,Tot,j} = \Delta M_{y,Tot,j} + \Delta M_{y,Cm,j} + M_{y,Tot,j+1} \quad (4-45)$$

In Eq.(4-44) $d_{T1,Arm,j}$, $d_{T2,Arm,j}$, $d_{T3,Arm,j}$ are the moment arms between the shear centre of the lifting surface bay and the forces acting about that centre and caused by the total inertia, aerodynamic loads, and the strut reaction, respectively.

4.8.4 Vector Transformations

In previous sections, moment increments are calculated in the local axes of the lifting surface (LS) bays, and they are summed from the tip of the LS towards the root to obtain the total moment distribution across the LS. This approach is correct if there is no load path change in the LS. The load path changes due to the sweep and dihedral angles causing products of the moment, which can be considered. This section will explain the transformation process using Euler's transformation matrix.

Eqs. (4-46),(4-47), and (4-48) show the rotation matrixes around x , y , and z axes with angles of β , α , and θ respectively.

$$R_x(\beta) = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\beta & -\sin\beta \\ 0 & \sin\beta & \cos\beta \end{bmatrix} \text{ for roll} \quad (4-46)$$

$$R_y(\alpha) = \begin{bmatrix} \cos\alpha & 0 & \sin\alpha \\ 0 & 1 & 0 \\ -\sin\alpha & 0 & \cos\alpha \end{bmatrix} \text{ for pitch} \quad (4-47)$$

$$R_z(\theta) = \begin{bmatrix} \cos\theta & -\sin\theta & 0 \\ \sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{bmatrix} \text{ for yaw} \quad (4-48)$$

The general rotation matrix, R , can be generated with the product of Eqs. (4-46)-(4-48) as given in Eq.(4-49).

$$[R] = R(z, \theta)R(y, \alpha)R(x, \beta) \quad (4-49)$$

$$= \begin{bmatrix} \cos\theta & -\sin\theta & 0 \\ \sin\theta & \cos\theta & 0 \\ 0 & 0 & 1 \end{bmatrix} \begin{bmatrix} \cos\alpha & 0 & \sin\alpha \\ 0 & 1 & 0 \\ -\sin\alpha & 0 & \cos\alpha \end{bmatrix} \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos\beta & -\sin\beta \\ 0 & \sin\beta & \cos\beta \end{bmatrix}$$

It should be noted that the matrix operation in Eq.(4-49) is not commutative, and the angle changes along the wing are applied around the x, y, and z axes, respectively. If the matrix operation in Eq.(4-49) is completed, Eq.(4-50) can be obtained.

$$[R] = \begin{bmatrix} \cos\theta\cos\alpha & \cos\theta\sin\alpha\sin\beta - \sin\theta\cos\beta & \cos\theta\sin\alpha\cos\beta + \sin\theta\sin\beta \\ \sin\theta\cos\alpha & \sin\theta\sin\alpha\sin\beta + \cos\theta\cos\beta & \sin\theta\sin\alpha\cos\beta - \cos\theta\sin\beta \\ -\sin\alpha & \cos\theta\sin\beta & \cos\alpha\cos\beta \end{bmatrix} \quad (4-50)$$

Having the 3D rotation matrix in Eq.(4-50), moment transformation from the local axes of the LS bays to the LS (wing) reference line can be obtained with Eq.(4-51).

$$\{\dot{M}_j\} = [R]\{M_j\} \quad (4-51)$$

In which,

$$\{\dot{M}_j\} = \begin{Bmatrix} \dot{M}_{x,j} \\ \dot{M}_{y,j} \\ \dot{M}_{z,j} \end{Bmatrix} \text{ and } \{M_j\} = \begin{Bmatrix} M_{x,j} \\ M_{y,j} \\ M_{z,j} \end{Bmatrix} \quad (4-52)$$

5 STRUCTURAL ANALYSIS AND SIZING

“Generally, a high degree of accuracy can be obtained only by using computer-based techniques, such as the finite element method. However, the simpler, quicker, and cheaper approximate methods can be used to advantage in the preliminary stages of design, when several possible structural alternatives are being investigated; they also provide an insight into the physical behaviour of structures which computer-based techniques do not” (Megson, 2013).

This chapter elucidates the structural principles and their implementation in the structural sizing of the components of the cantilever, strut- and truss-braced wings, and the analytical structural analysis methods to compute the internal loads acting on the structural components of a wing.

In Chapter 3, the simplified geometric models of the wings, struts, juries and offsets were derived and introduced. The types of loads arising from various flight scenarios are discussed in Chapter 4. In this chapter, the derivation of the analytical solutions to computing the critical stresses driving the sizing of the structural components of a wing will be explained. The sizing criteria of those components considering the requirements of CS-25 (EASA, 2020) for limit and ultimate loads are discussed in 4.1.2 Structural Sizing Criteria. The structures are sized to resist the bending and shear stresses as well as the critical stresses for column buckling, local buckling and panel buckling generated by the ultimate and limit loads. The bird strike criteria are also used to size the leading and trailing edges of the juries, strut, and offset.

Moreover, simplified geometric models are developed to compute the internal loads acting on the strut and the juries. Two analytical solutions are derived to compute the strut reaction forces using Castigliano’s theorem in the simplified indeterminate TBW models. The first model is a simplified TBW configuration with one concentrated load. In the second model, the simplified configuration is modelled with unlimited concentrated loads. Direct analytical solutions are presented to compute the strut reaction force in both models. The simplified jury model presents a conservative approach to computing the internal axial loads acting on the juries. Finally, these internal loads are used to size the load-carrying structure of the strut and juries.

5.1 Wing Box Stress Analysis and Sizing

Single-cell wing boxes are commonly used in subsonic or transonic aircraft with a moderate or a HARW. The geometric model for this type of wing box is explained in Chapter - 3, and in this section, the computations of the stresses on their structural elements are presented.

5.1.1 Structural Idealisation

Stress analysis of aircraft components is complicated and requires high computational power and long computational time, which are not desired at the conceptual or preliminary design stage. Hence, structural idealisation is used to simplify the geometry of aircraft components to provide a sufficient accuracy level and short computational time. This section explains the procedure for the structural idealisation of the wing box.

Assumptions for the idealisation,

- The cross-sectional areas of the stringers and spar caps are smaller than those of all sections. Hence the stress variation along the cross-section of a stringer or a flange is small due to the bending stress. It can be assumed that the direct stress along the cross-section of a stringer or a spar cap is constant.
- The distance between the centroid of the stringers and the adjacent skin is small; hence their centroid can be assumed as overlapped.
- Following the theory described in (Megson, 2013), it is assumed that the direct stresses are only carried by the mass booms while the shear stress is carried only by the wing box skin. Hence the amount of direct stress the skin carries can be added to the mass-boom areas.

Following the assumptions, we can change stringer and spar caps with simpler mass booms whose centroid coincides with the centre of the skin and the cross-sectional areas are the same as that of the relevant stringers or caps as previously presented in Figure 17 and Figure 18. The shape of the wing box cross-section is also idealised, as shown in Figure 18. An irregular hexagonal geometry is chosen to represent the actual wing box better.

5.1.2 Stress due to Bending Moment

The axial stress in i th mass boom of the j th section (depicted in Figure 48) can be written as,

$$\sigma_{j,i} = -\frac{M_{x,j}}{I_{xx,j}} z_{j,i} + \frac{M_{z,j}}{I_{zz,j}} x_{j,i} + \frac{F_{y,j}}{N_{boom,j} * B_{j,i}} \quad (5-1)$$

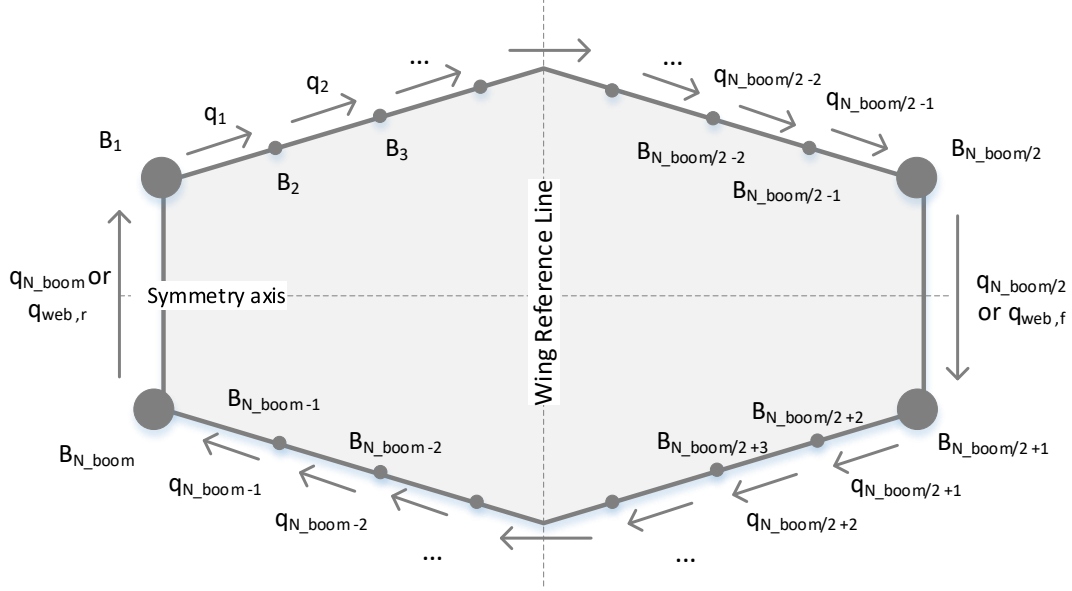


Figure 48 Representation of the shear flows between the mass booms of the idealised wing box section.

In an actual aircraft wing box, the cross-section areas of the stringers are usually equal in j th wing box section and can be characterised with B_j for simplification instead of $B_{j,i}$. In addition to that, the cross-section area of the spar caps, $B_{C,j}$, is usually different than that of the stringers, B_j , and to include this difference in the model, a parameter of SCR is introduced.

$$SCR_j = \frac{B_{C,j}}{B_j} \quad (5-2)$$

If the $B_{C,j}$ is written in terms of B_j , the second moment of area of j th wing box section around the x and z axes can be formulated as,

$$I_{xx,j} = 2 * B_j * \left(\sum_{i=2}^{\frac{N_{Boom}-1}{2}} z_i^2 + SCR * z_1^2 + SCR * z_{N_{Boom}/2}^2 \right) \quad (5-3)$$

$$I_{zz,j} = 2 * B_j * \left(\sum_{i=2}^{\frac{N_{Boom}-1}{2}} x_{j,i}^2 + SCR * x_{j,1}^2 + SCR * x_{j,N_{Boom}/2}^2 \right) \quad (5-4)$$

in which $N_{Boom}/2$ is the total mass boom count on the one surface of a wing and $x_{j,i}$ and $y_{j,i}$ are the distances of the mass booms to the symmetry axes of the wing box section for stringers and the spar caps. If Eqs.(5-3) and (5-4) are substituted in Eq.(5-1) and B_j is isolated then Eq.(5-5) can be derived to compute B_j .

$$B_j = \left| \left(- \frac{M_{x,j}}{2 \left(\sum_{j=2}^{\frac{N_{Boom}}{2}-1} z_j^2 + SCR * z_1^2 + SCR * z_{N_{Boom}/2}^2 \right)} z_i \right. \right. \\ \left. \left. + \frac{M_{z,j}}{2 \left(\sum_{j=2}^{\frac{N_{Boom}}{2}-1} x_j^2 + SCR * x_1^2 + SCR * x_{N_{Boom}/2}^2 \right)} x_i \right. \right. \\ \left. \left. + \frac{F_{y,j}}{(N_{boom} - 4 + 4SCR)} \right) \frac{1}{\sigma_{Y,i}} \right| \quad (5-5)$$

In which B_j is the cross-section area of any stringer, and $\sigma_{Y,i}$ is the yield stress of the mass boom material in the j th wing box section. As previously discussed in 4.1.2 Structural Sizing Criteria, for composite materials, $\sigma_{U,i}$ will be used instead of $\sigma_{Y,i}$ and ultimate loads will be applied instead of limit loads. Yielding stress ($\sigma_{Y,i}$) is only used for the isotropic materials under limit loads conditions.

5.1.3 Shear Flow

The shear flow occurred on the i th skin panel of the j th idealised wing box sections is presented with $q_{j,i}$ which can be written as q_i for a particular wing box section for simplicity, illustrated in Figure 48. The shear flow is accepted positive in the clockwise direction, as shown in Figure 48 for a single-cell wing box section.

From (Megson, 2013), “the alternative method” was chosen to calculate the shear flow between the mass booms. The theory of the method is well-explained in the reference. This section will explain its application for the hexagonal wing box model. The shear flow between two mass booms, as illustrated in Figure 48, can be calculated with Eq.(5-6).

$$q_i - q_{i-1} = -\partial P_i / \partial y \quad (5-6)$$

Where the P_i is the direct load in the i th mass boom. As explained (Megson, 2013), for a constant shear force over a wing bay, Eq.(5-6) can be rearranged as,

$$q_i - q_{i-1} = -\Delta P_i \quad (5-7)$$

In which $-\Delta P_i$ is the axial load variation along a unit length of a boom. In the case of non-constant shear force over a wing bay, the method provides an approximate solution. If the shear flow of one skin panel is known, the rest of the unknown shear flows can be calculated with Eq.(5-8), which is obtained from Eq.(5-7).

$$q_i = -\Delta P_i + q_{i-1} \quad (5-8)$$

Eq.(5-8) can be rearranged as shown in (5-9) to write all the shear flows in terms of the shear flow of the first skin panel, q_1 , which is between mass boom-1 and mass boom-2.

$$\begin{aligned} q_2 &= -\Delta P_2 + q_1 \\ q_3 &= -\Delta P_2 + q_1 - \Delta P_3 \\ q_4 &= -\Delta P_4 + q_1 - \Delta P_4 - \Delta P_5 \\ &\dots \end{aligned} \quad (5-9)$$

The set of equations above can be written with the sum symbol as given in Eq.(5-10).

$$q_i = q_1 - \sum_{a=2}^i \Delta P_a \quad (5-10)$$

In Eq.(5-10) there are two unknowns of q_i and q_1 and one equation. To solve the equation, another equation is required, and it can be derived using the moment equilibrium around the wing box centreline as given in Eq.(5-11).

$$M_y + \sum_{i=1}^{N_{boom}} (q_i) d_i l_i = 0 \quad (5-11)$$

In which d_i is the moment arm of the shear flow on a relevant skin panel and l_i is the length of each panel which is presented in Figure 49 and will be explained in more detail below. Eq.(5-12) can be derived by substituting Eq.(5-10) in Eq.(5-11),

$$M_y + \sum_{i=1}^{N_{boom}} \left(q_1 - \sum_{a=2}^i \Delta P_a \right) d_i l_i = 0 \quad (5-12)$$

If q_1 in (5-12) is isolated as given in Eq.(5-13),

$$M_y + \sum_{i=1}^{N_{boom}} (q_1) d_i l_i - \sum_{i=2}^{N_{boom}} \left(\sum_{a=2}^i \Delta P_a \right) d_i l_i = 0 \quad (5-13)$$

the formulae in Eq.(5-14) can be derived to calculate q_1 .

$$q_1 = \frac{\sum_{i=2}^{N_{boom}} \left(\sum_{a=2}^i \Delta P_a \right) d_i l_i - M_y}{\sum_{i=1}^{N_{boom}} d_i l_i} \quad (5-14)$$

When the unknown shear flows in the first skin panel, q_1 , is calculated with Eq.(5-14). The equations in (5-9) can be solved step by step to calculate other shear flows in the rest of the skin panels.

In Eq.(5-11), d_i is the moment arm of each shear flow on a particular skin panel, and it can be derived as Eq.(5-15) using the geometric relations in Figure 49, except the moment arms of the spar webs, which are $d_{N_{boom}/2} = d_{N_{boom}} = w_{WB}/2$.

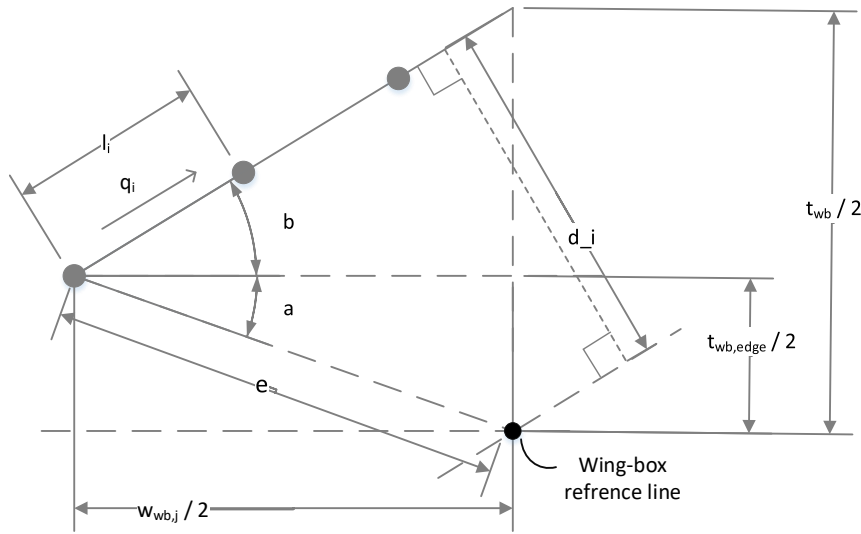


Figure 49 Geometric model to calculate the moment arm of skin shear flow.

$$d_i = \sin(a + b)e \quad (5-15)$$

In which a , b , and e can be computed with Eqs. (5-16)-(5-18).

$$a = \arctan(t_{wb,edge}/w_{wb}) \quad (5-16)$$

$$b = \arctan(t_{wb} - t_{wb,edge})/w_{wb} \quad (5-17)$$

$$e = \sqrt{(w_{wb}/2)^2 + (t_{wb,edge}/2)^2} \quad (5-18)$$

The formulation in Eq.(5-14) can be applied to all wing sections, and the notation of shear flow can be represented with $q_{j,i}$ as explained previously. Hence in each wing box panel along the wing, the material thickness to resist the shear flow can be calculated with Eq.(5-19).

$$t_{shear,j,i} \geq \frac{q_{j,i}}{\sigma_{Y,60\%,j,i}} \quad (5-19)$$

In which $\sigma_{Y,60\%,j,i}$ is the shear yield stress of i th panel materials in the j th wing box section. As previously discussed in 4.1.2 Structural Sizing Criteria, for composite materials, $\sigma_{U,60\%,j,i}$ will be used instead of $\sigma_{Y,60\%,j,i}$ and ultimate loads will be applied instead of limit loads. Yielding stress ($\sigma_{Y,60\%,j,i}$) is only used for the isotropic materials under limit loads conditions.

5.1.4 Panel Buckling

Most aircraft structures consist of thin plates stiffened by the stringers and prone to failure at critical buckling stress. For these structural elements, buckling is a crucial failure mode, and its computation is essential in aircraft design. This section will explain the analysis procedure to calculate the buckling stresses of the simply supported panels and will provide the minimal element thickness to avoid these phenomena.

The minimum thickness of the skin panel between the booms can be calculated for elastic buckling constrain with Eq.(5-20).

$$t_{E.Buckling,j,i} = q_{j,i}/\tau_{cr,e} \quad (5-20)$$

$q_{j,i}$ is calculated with Eq.(5-10) whereas $\tau_{cr,e}$ is critical elastic buckling shear stress for an isotropic thin panel, which can be obtained from the reference (Mr M.E. Grayley, 1995) as,

$$\tau_{cr,E} = K \times E \times (t_{E.Buckling,j,i}/l_{j,i})^2 \quad (5-21)$$

K is given in a graph (Mr M.E. Grayley, 1995). For a panel with all edges simply supported, an equation can be derived for the term K from the reference by fitting a curve as in Eq. (5-22).

$$K = 3.4(l_{j,i}/b_{Bay,j})^2 + 5 \quad (5-22)$$

Hence substituting Eq.(5-21) and Eq.(5-22) in Eq.(5-20) $t_{E. Buckling,j,i}$ can be calculated with Eq.(5-23).

$$t_{E. Buckling,j,i} \geq \sqrt[3]{q_{j,i} \times l_{j,i}^2 / \left(\left(3.4 \times \left(\frac{l_{j,i}}{b_{bay,j}} \right)^2 + 5 \right) \times E \right)} \quad (5-23)$$

The equation can be corrected for plasticity with the factor η_B from (Mr M.E. Grayley, 1995). Therefore, the shear stress at which the plates buckle, τ_{cr} , can be calculated by $\tau_{cr} = \eta_B * \tau_{cr,e}$. If Eq.(5-23) is rearranged for plasticity reduction, minimum thickness to avoid buckling, $t_{E. Buckling,j,i}$, can be found by Eq.(5-24).

$$t_{Buckling,j,i} \geq \sqrt[3]{q_{j,i} \times l_{j,i}^2 / \left(\left(3.4 \times \left(\frac{l_{j,i}}{b_{bay,j}} \right)^2 + 5 \right) * E * \eta_B \right)} \quad (5-24)$$

It should be noted that the method to compute the minimum panel thickness in this section is primarily developed for isotropic materials and it is also used for quasi-isotropic composite materials to simplify computation considering their similarity to the isotropic materials. Unless quasi-isotropic composite materials are used, the local buckling methods presented in this section will be less applicable and more detailed methods are suggested to be used.

5.1.5 Global Buckling

The wing box structures are idealized using the methods from Ref. (Megson, 2013). The idealization method is not suitable for the realistic global buckling analysis of the skin panels as it assumes that the bending stresses are only carried by the mass booms and the shear stresses are only resisted by the skin panels.

The global buckling of the wing panels is assumed to be prevented by sufficient rib pitching while the global buckling of the strut, jury and offset are considered constraints in sizing as discussed in the following sections. The analytical ME of the required number of ribs in the preliminary design stage, according to Torenbeek (1992), is impractical because it necessitates a thorough wing structural design, hence the mass of the ribs is covered by semi-empirical equations.

The inclusion of the global buckling as a constraint might increase the wing mass if this phenomenon occurs under the studied loading cases. A specific consideration of this constraint in the inner part of the SBW (between the strut attachment and wing root) should be studied in future studies. The validity of the assumptions can be seen from the validation and verification studies of this work; however, the detailed effect of these assumptions together with different stiffener shapes on the developed method accuracy can be studied in further studies if the sensitivity level of the model is aimed to be improved.

5.2 Structural Sizing Procedure

The details of the sizing of the wing box skin panels and mass booms are explained in this section.

5.2.1 Shear Resistant Material

Eqs. (5-19) and (5-24) can provide the thinnest wing box skin panel to resist the shear or critical buckling stresses in one loading case. The required thickness of the panel for one loading case should be the thickest one of $t_{Shear,j,i}$ and $t_{Buckling,j,i}$ as given in (5-25).

$$t_{Panel,j,i,n} = \max(t_{Shear,j,i}, t_{Buckling,j,i}) \quad (5-25)$$

If the same process is repeated for each loading case presented in Chapter 4, then a total of N_{Case} thickness value will be obtained. The final size of the studied panel will then be the maxima in this thickness list, as given in Eq. (5-26).

$$t_{Panel,j,i} = \max(\{t_{Panel,j,i,n} : n = 1, \dots, N_{Case}\}) \quad (5-26)$$

Eq.(5-26) results in different panel thicknesses for each of the panels along the wing; however, in the real-life application, the panel thicknesses in the same wing box section might be equal due to manufacturing difficulties and complexity caused by the shear lags except the spar webs. Considering these design requirements, Eqs. (5-27) and (5-28) can be used to find the panel thickness in the j th wing box section for the upper, $t_{Panel\ section,u,j}$, and the lower, $t_{Panel\ section,l,j}$, surfaces respectively while the front, $t_{Web\ front,j}$, and the rear, $t_{Web\ rear,j}$, spar web thicknesses can be obtained from Eqs.(5-29) and (5-30), respectively.

$$t_{Panel\ section,u,j} = \max\left(\left\{t_{Panel,j,i} : i = 1, \dots, \frac{N_{Boom,j}}{2} - 1\right\}\right) \quad (5-27)$$

$$t_{Panel\ section,l,j} = \max \left(\left\{ t_{Panel,l,j,i} : i = \frac{N_{Boom,j}}{2} - 1, \dots, N_{Boom,j} - 1 \right\} \right) \quad (5-28)$$

$$t_{Web\ front,j} = \left\{ t_{Panel\ cases,j,i} : i = \frac{N_{Boom,j}}{2} \right\} \quad (5-29)$$

$$t_{Web\ rear,j} = \left\{ t_{Panel\ cases,j,i} : i = N_{Boom,j} \right\} \quad (5-30)$$

The above-given shear resistance panel sizing method will be labelled with NEW-2. Although NEW-2 is chosen for this study, another sizing approach (NEW-1) is also tested, and its results are also used in the validation studies. NEW-1 differs from the first one with its skin panel sizing method. Instead of selecting the same panel thickness for the same wing box section of the same surface, each panel is assumed to have a different thickness. Hence from N_{Case} loading cases and $N_{Sections}$ wing sections, for example, $N_{Case} * N_{Sections} * N_{Panel}$ panel thickness values are recorded for N_{Panel} panel number. Each panel is sized for its worst loading cases regardless of its wing box section. The validation section will discuss the comparison of the NEW-1 approach to NEW-2.

5.2.2 Bending Resistant Material

As explained in the structural idealisation section, axial stresses caused by the bending moments are carried by the mass booms, which are stringers and the spar caps in a wing box. In Section 5.1.2 Stress due to Bending Moment, the calculation process for the cross-section area of a stringer in j th wing box section is explained and Eq.(5-5) can be used for this purpose. The cross-section area of the spar caps in j th wing box section then can be computed with Eq.(5-31).

$$B_{Cap,j} = B_j * SCR \quad (5-31)$$

5.3 Solutions to the Indeterminate SBW Problem

At the preliminary design stage, the strut reaction force has been defined as user input, or it has been calculated by applying beam equations on simplified SBW models in the literature, as seen in references (Park, 1980; Chiozzotto, 2016). Castigliano's second theorem has been used in previous studies at Cranfield University as an alternative to beam equations.

The advantage of using beam equations or Castigliano's approach is the fast computation time for a direct solution compared to the higher fidelity methods such as

finite element analysis (FEA). FEA can increase the complexity and computational time at each loading case, and it is likely to significantly increase the mathematical model's total computational time (MM). Therefore, analytical solutions are employed at the conceptual or preliminary design stage as they can provide direct and rapid solutions.

The analytical solutions in the literature are generally for the models with one concentrated load, as shown in Figure 50. If the concentrated load numbers are increased and the locations of the concentrated loads are changed, the analytical solutions usually become invalid in those models, and new solutions are required. This approach requires a set of analytical solutions for each simplified model with a different number of concentrated loads. This requirement limits the design space exploration of some configurations when the configuration especially benefits from the distributed propulsion technologies.

Hence, to study the distributed propulsion TBW configurations, a general analytical solution with an unlimited number of concentrated loads was required. The author created a geometric model, as illustrated in Figure 51 and derived a general analytical solution for strut reaction force in a TBW configuration with an unlimited number of concentrated loads. This solution enables studying the TBW aircraft with distributed propulsion technologies. This section will present two solutions to strut reaction force calculation using Castigliano's theorem. One model is created with one concentrated load, shown in Figure 50, and the other model has an unlimited number of concentrated loads, shown in Figure 51. The results of the models are validated by comparing their results with FEA results, which will be discussed in Section - 5.3.4.

5.3.1 Simplifications, Assumptions and Their Effects

To apply Castigliano's theorem with less complexity, the TBW configuration is simplified with the assumptions listed below. Some of these assumptions are similar to those in the literature (Park, 1980; Chiozzotto, 2016).

- The model has distributed and concentrated loads. Concentrated loads are located at the wing box reference line.
- The distributed loads are modelled using the triangular distribution. The total distributed load magnitude and shape calculated in the loading action module are redistributed in a triangular shape with the same magnitude to be used in this section.

- The beam is assumed to be uniform and straight between the wing root and the strut-wing attachment point.

These simplifications are likely to affect the accuracy of the model. The output of the models presented in this section is the reaction force, R_s . The value of R_s then is used in the load analysis, as explained before. Therefore, the accuracy of R_s will affect the complete model accuracy for TBW. The verification of this approach is achieved by comparing the wing ME results of the new MM with the ME of TBW aircraft from a more detailed study (Bradley, Allen and Droney, 2015), and it will be discussed in Section - 6.5.3 Validation of the Mass Estimations. The close results verified the accuracy of the approaches presented in this section. In addition, parametric studies with TBW also increase the credibility of the approaches. However, more detailed studies to investigate the effects of these assumptions on the strut reaction force are suggested to study in future works.

In addition to the above assumptions, another assumption (zero deflection) is made when deriving a direct solution using Castigliano's theorem. The effect of this assumption is explained by referring to (Park, 1980) below. The general equation of Castigliano's second theorem using the energy approach can be written as in Eq. (5-32) to compute the deflection at the strut joint, Δ_s .

$$\Delta_s = \int_0^{b^*} M \left(\frac{\partial M}{\partial R_s} \right) \frac{dx}{EI} \quad (5-32)$$

In Eq. (5-32), M is the moment equation along the wing and R_s are the reaction applied by the strut to the wing, EI is the flexural rigidity of the beam.

If the deflection of the wing at the strut joint, Δ_s is assumed to be zero, Eq. (5-32) can be rearranged to compute the strut reaction force R_s in the simplified indeterminate TBW problem. The effect of the assumption of zero deflection is previously discussed in (Park, 1980), and its validity is explained for computing the strut reaction force.

5.3.2 TBW Model with One Concentrated Load

The simplified TBW model with one concentrated load is shown in Figure 50. Equation (5-32) can be rearranged for this model as given in Eq. (5-33).

$$(\Delta_c) = \int_0^{l_1} M_1(0) \frac{dx_1}{EI} + \int_0^{l_2} M_2 \left(\frac{\partial M_2}{\partial R_s} \right) \frac{dx_2}{EI} + \int_0^{l_3} M_3 \left(\frac{\partial M_3}{\partial R_s} \right) \frac{dx_3}{EI} \quad (5-33)$$

In Eq. (5-33), the moment equations should be derived for each section shown in Figure 50. The sections are between the fuselage and the engine, the engine and the strut, and the wing tip. First, the total lift load generated by each section should be calculated to derive the moment equations.

To compute the total lifts, we can start calculating the distributed loads w_1, w_2, w_3 for x_1, x_2, x_3 in the described sections. The formulations to compute the magnitude of the distributed loads of w_1, w_2, w_3 at the sections given in Eq. (5-34).

$$W_1 = \frac{x_1 W}{l}$$

$$W_2 = \frac{(l_1 + x_2)(W)}{l}$$

$$W_3 = \frac{(l_e + x_3)W}{l}$$
(5-34)

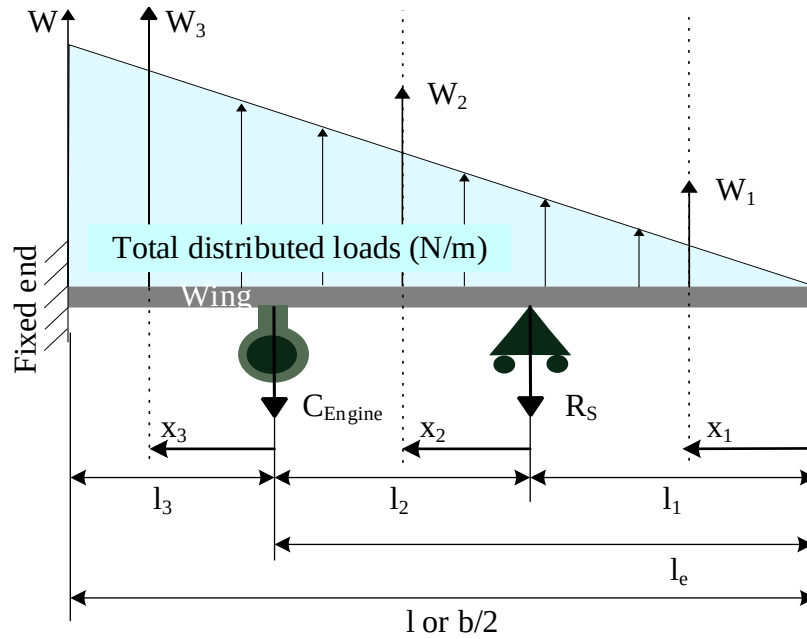


Figure 50 Figure 1 Illustration of the simplified TBW model for strut reaction force computation.

After computing W_1, W_2, W_3 , the formulations in Eq.(5-35) can be used to compute the total lift loads of L_1, L_2, L_3 at each wing section.

$$\begin{aligned}
 L_1 &= \frac{x_1^2 W}{2l} \\
 L_2 &= \frac{(l_1^2 + 2l_1 x_2 + x_2^2) W}{2l} \\
 L_3 &= \frac{W(l_e^2 + 2l_e x_3 + x_3^2)}{2l}
 \end{aligned} \tag{5-35}$$

Using the formulations in Eqs. (5-34) and (5-35), the moment equations of M_1, M_2, M_3 for each wing section at x_1, x_2, x_3 , respectively, can be computed using formulations in Eq. (5-36).

$$\begin{aligned}
 M_1 &= \frac{(x_1)^3 W}{6l} \\
 M_2 &= \frac{(Wl_1^3 + 3Wl_1^2 x_2 + 3Wl_1 x_2^2 + Wx_2^3)}{6l} - R_S x_2 \\
 M_3 &= \frac{Wl_e^3 + 3Wl_e^2 x_3 + 3Wl_e x_3^2 + Wx_3^3}{6l} - C_{Engine} x_3 - R_S (l_2 + x_3)
 \end{aligned} \tag{5-36}$$

In Eq. (5-36), C_{Engine} is inertia load applied by the engine. Using the moment equations in Eq. (5-36), the terms of $\left(\frac{\partial M_1}{\partial R_S}\right)$, $\left(\frac{\partial M_2}{\partial R_S}\right)$, $\left(\frac{\partial M_3}{\partial R_S}\right)$ in Eq. (5-33) can be derived as given in Eq. (5-37).

$$\begin{aligned}
 \frac{\partial M_1}{\partial R_S} &= 0 \\
 \frac{\partial M_2}{\partial R_S} &= -x_2 \\
 \frac{\partial M_3}{\partial R_S} &= -l_2 - x_3
 \end{aligned} \tag{5-37}$$

If Eqs. (5-37) and Eq. (5-36) are substituted in Eq. (5-33), Eq.(5-38) can be derived.

$$\begin{aligned}
(\Delta_c) = & \int_0^{l_1} \frac{(x_1)^3 W}{6l} (0) \frac{dx_1}{EI} \\
& + \int_0^{l_2} \left(\frac{(Wl_1^3 + 3Wl_1^2 x_2 + 3Wl_1 x_2^2 + Wx_2^3)}{6l} - R_S x_2 \right) (-x_2) \frac{dx_2}{EI} \\
& + \int_0^{l_3} \left(\frac{(Wl_e^3 + 3Wl_e^2 x_3 + 3Wl_e x_3^2 + Wx_3^3)}{6l} - C_{Engine} x_3 - R_S l_2 \right. \\
& \left. - R_S x_3 \right) (-l_2 - x_3) \frac{dx_3}{EI}
\end{aligned} \tag{5-38}$$

If the previously explained assumption of zero deflection at the strut-wing joint is applied to Eq. (5-38), all the integrals are taken and the term R_S is isolated, Eq. (5-39) can be derived. Equation (5-39) is a direct analytical solution for the computation of the strut reaction force at the wing strut-wing attachment point, as illustrated in Figure 50.

$$\begin{aligned}
R_S = & \frac{6 \left\{ -\frac{(Wl_1^3 l_2^2)}{12l} - \frac{(Wl_1^2 l_2^3)}{6l} - \frac{(Wl_1 l_2^4)}{8l} - \frac{(Wl_2^5)}{30l} - \frac{Wl_e^3 l_2 l_3}{6l} - \frac{Wl_e^2 l_3^2 l_2}{4l} - \frac{Wl_e l_3^3 l_2}{6l} \right.}{-(2l_2^3 + 6l_3 l_2^2 + 6l_3^2 l_2 + 2l_3^3)} \\
& \left. - \frac{Wl_3^4 l_2}{24l} + \frac{C_{Engine} l_3^2 l_2}{2} - \frac{Wl_e^3 l_3^2}{12l} - \frac{Wl_e^2 l_3^3}{6l} - \frac{Wl_e l_3^4}{8l} - \frac{Wl_3^5}{30l} + \frac{C_{Engine} l_3^3}{3} \right\}}{-(2l_2^3 + 6l_3 l_2^2 + 6l_3^2 l_2 + 2l_3^3)}
\end{aligned} \tag{5-39}$$

The validation of the result of Eq. (5-39) for the simplified model in Figure 50 is presented in Section - 5.3.4.

5.3.3 Unlimited Number of Concentrated Loads

In this study, a simplified SBW model with unlimited concentrated loads was created, as illustrated in Figure 51. Equation (5-32) can be rearranged considering the concentrated loads located at the inner and outer parts of the wing, as given in Eq. (5-40).

$$\begin{aligned}
\Delta_s = & \int_0^{l_{0,1}} M_{0,1} \left(\frac{\partial M_{0,1}}{\partial R_S} \right) \frac{dx}{EI} \dots + \int_{l_{0,N}}^{l_s} M_{0,N} \left(\frac{\partial M_{0,N}}{\partial R_S} \right) \frac{dx}{EI} + \int_{l_s}^{l_2} M_1 \left(\frac{\partial M_1}{\partial R_S} \right) \frac{dx}{EI} \dots \\
& + \int_{l_N}^{b^*} M_N \left(\frac{\partial M_N}{\partial R_S} \right) \frac{dx}{EI}
\end{aligned} \tag{5-40}$$

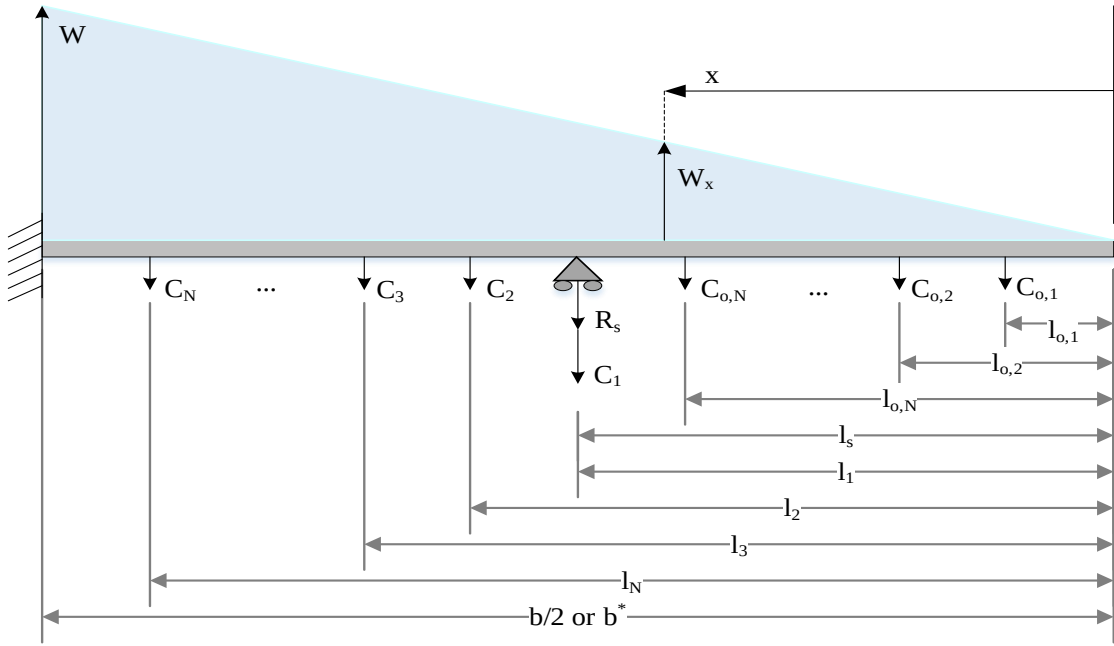


Figure 51 Simplified indeterminate SBW wing model with triangular distributed and unlimited concentrated loads.

In Eq. (5-40), $M_{0,1}, \dots, M_{0,N}$ are the moment equations between the concentrated loads of $C_{0,1}$ and $C_{0,N}$. And $M_1, \dots, M_N$ are the moment equations between the concentrated loads of C_1 and C_N . These equations are derived step by step below. The value of unit per length distributed load at any position along the wing, as shown in Figure 51, can be calculated with Eq.(5-41).

$$W_x = \frac{Wx}{b^*} \quad (5-41)$$

The total distributed load, L_x between the wing tip and at the x position along the wing can be written as in Eq.(5-42).

$$L_x = \frac{x^2 W}{2b^*} \quad (5-42)$$

The moment equations for N number of concentrated loads located at the outer part of the wing between the strut attachment and the wing tip can be written with the set of equations in Eq. (5-43).

$$\text{For } [0, l_{o,1}], \quad M_{o,1} = \frac{x^3 W}{6b^*}$$

$$\text{For } (l_{o,1}, l_{o,2}], \quad M_{o,2} = \frac{x^3 W}{6b^*} - C_{o,1}(x - l_{o,1}) \quad (5-43)$$

...

$$\begin{aligned} \text{For } (l_{o,N}, l_s], \quad M_{o,N} \\ = \frac{x^3 W}{6b^*} - C_{o,1}(x - l_{o,1}) - C_{o,2}(x - l_{o,2}) \dots - C_{o,N}(x - l_{o,N}) \end{aligned}$$

The moment equations for N number of concentrated loads located at the inner section of the wing between the strut attachment and the wing root can be written with the set of equations in (5-44).

$$\begin{aligned} \text{For } (l_s, l_2], \quad M_1 \\ = \frac{x^3 W}{6b^*} - C_{o,1}(x - l_{o,1}) - C_{o,2}(x - l_{o,2}) \dots \\ - C_{o,N}(x - l_{o,N}) - R_s(x - l_s) - C_1(x - l_1) \end{aligned} \quad (5-44)$$

...

$$\begin{aligned} \text{For } (l_N, b^*], \quad M_N \\ = \frac{x^3 W}{6b^*} - C_{o,1}(x - l_{o,1}) - C_{o,2}(x - l_{o,2}) \dots \\ - C_{o,N}(x - l_{o,N}) - R_s(x - l_s) - C_1(x - l_1) \\ - C_2(x - l_N) \dots - C_N(x - l_N) \end{aligned}$$

If the moment equations in Eqs. (5-43) and (5-44) are substituted in (5-40), it can be seen that the partial derivative of the moment equations for the outer section will be zero as they are not dependent to R_s . Hence the Eq.(5-40) can be rearranged as in Eq.(5-45).

$$\Delta_s = \int_{l_s}^{l_2} M_1 \left(\frac{\partial M_1}{\partial R_s} \right) \frac{dx}{EI} \dots + \int_{l_N}^{b^*} M_N \left(\frac{\partial M_N}{\partial R_s} \right) \frac{dx}{EI} \quad (5-45)$$

The derivatives of the inner section moment equations for ∂R_s are equal to $(l_s - x)$. If these derivatives are substituted into Eq.(5-45), Eq.(5-46) can be derived.

$$\Delta_s = \int_{l_s}^{l_2} M_1 (l_s - x) \frac{dx}{EI} + \dots \int_{l_N}^{b^*} M_N (l_s - x) \frac{dx}{EI} \quad (5-46)$$

If the moment equations from Eq. (5-43) and Eq. (5-44) are substituted into Eq.(5-45). The first integral between $(l_s, l_2]$ can be written as,

$$\begin{aligned}
& \int_{l_s}^{l_2} M_1(l_s - x) \frac{dx}{EI} \quad (5-47) \\
&= l_s \left[\frac{x^4 W}{24b^*} - \frac{x^2 C_{o,1}}{2} + x C_{o,1} l_{o,1} \dots - \frac{x^2}{2} C_{o,N} + x C_{o,N} l_{o,N} - \frac{x^2}{2} R_s \right. \\
&\quad \left. + x R_s l_s - \frac{x^2}{2} C_1 - x C_1 l_1 \right]_{l_s}^{l_2} \\
&\quad - \left[\frac{x^5 W}{30b^*} - \frac{x^3 C_{o,1}}{3} + \frac{x^2}{2} C_{o,1} l_{o,1} \dots - \frac{x^3}{3} C_{o,N} + \frac{x^2}{2} C_{o,N} l_{o,N} - \frac{x^3}{3} R_s \right. \\
&\quad \left. + \frac{x^2}{2} R_s l_s - \frac{x^3}{3} C_1 + \frac{x^2}{2} C_1 l_1 \right]_s^{l_2}
\end{aligned}$$

To better organise Eq.(5-47), A and B can be introduced as given in Eqs. (5-48) and (5-49),

$$\begin{aligned}
A = & \frac{x^4 W}{24b^*} - \frac{x^2 C_{o,1}}{2} + x C_{o,1} l_{o,1} \dots - \frac{x^2}{2} C_{o,N} + x C_{o,N} l_{o,N} - \frac{x^2}{2} R_s + x R_s l_s - \frac{x^2}{2} C_1 \quad (5-48) \\
& - x C_1 l_1
\end{aligned}$$

$$\begin{aligned}
B = & \frac{x^5 W}{30b^*} - \frac{x^3 C_{o,1}}{3} + \frac{x^2}{2} C_{o,1} l_{o,1} \dots - \frac{x^3}{3} C_{o,N} + \frac{x^2}{2} C_{o,N} l_{o,N} - \frac{x^3}{3} R_s + \frac{x^2}{2} R_s l_s \quad (5-49) \\
& - \frac{x^3}{3} C_1 + \frac{x^2}{2} C_1 l_1
\end{aligned}$$

Similarly, all the integrals in Eq.(5-46) can be expanded, and the equations can be written as in Eq. (5-50).

$$\begin{aligned}
& \text{For } (l_s, l_2], \quad \int_{l_2}^{l_3} M_2(l_s - x) \frac{dx}{EI} \quad (5-50) \\
&= l_s \left[A - \frac{x^2}{2} C_2 + x C_2 l_2 \right]_{l_2}^{l_3} - \left[B - \frac{x^3}{3} C_2 + \frac{x^2}{2} C_2 l_2 \right]_{l_2}^{l_3}
\end{aligned}$$

...

$$\begin{aligned}
\text{For } (l_N, b^*), \quad & \int_{l_N}^{b^*} M_N(l_s - x) \frac{dx}{EI} \\
&= l_s \left[A - \frac{x^2}{2} C_2 + x C_2 l_2 - \dots - \frac{x^2}{2} C_N + x C_N l_N \right]_{l_N}^{b^*} \\
&\quad - \left[B - \frac{x^3}{3} C_2 + \frac{x^2}{2} C_2 l_2 - \dots - \frac{x^3}{3} C_N + \frac{x^2}{2} C_N l_N \right]_{l_N}^{b^*}
\end{aligned}$$

It can be seen from the series of equations in Eq.(5-50) that the same parts of A and B of the equations will remove each other at the section boundaries, and finally, Eq.(5-46) becomes as follows in Eq. (5-51).

$$\begin{aligned}
\Delta_s = & -l_s \left[\frac{l_s^4 W}{24b^*} - \frac{l_s^2 C_{o,1}}{2} + l_s C_{o,1} l_{o,1} \dots - \frac{l_s^2}{2} C_{o,N} + l_s C_{o,N} l_{o,N} - \frac{l_s^2}{2} R_s + l_s R_s l_s \right] \quad (5-51) \\
& + \left[\frac{l_s^5 W}{30b^*} - \frac{l_s^3 C_{o,1}}{3} + \frac{l_s^2}{2} C_{o,1} l_{o,1} \dots - \frac{l_s^3}{3} C_{o,N} + \frac{l_s^2}{2} C_{o,N} l_{o,N} \right. \\
& \quad \left. - \frac{l_s^3}{3} R_s + \frac{l_s^2}{2} R_s l_s \right] \\
& + l_s \left[\frac{l_1^2}{2} C_1 - l_1 C_1 l_1 \right] - \left[\frac{l_1^3}{3} C_1 - \frac{l_1^2}{2} C_1 l_1 \right] \\
& + l_s \left[\frac{l_2^2}{2} C_2 - l_2 C_2 l_2 \right] - \left[\frac{l_2^3}{3} C_2 - \frac{l_2^2}{2} C_2 l_2 \right] \\
& \dots \\
& + l_s \left[\frac{l_N^2}{2} C_N - l_N C_N l_N \right] - \left[\frac{l_N^3}{3} C_N - \frac{l_N^2}{2} C_N l_N \right] \\
& + l_s \left[\frac{b^{*4} W}{24b^*} - \frac{b^{*2} C_{o,1}}{2} + b^* C_{o,1} l_{o,1} \dots - \frac{b^{*2}}{2} C_{o,N} + b^* C_{o,N} l_{o,N} - \frac{b^{*2}}{2} R_s + b^* R_s l_s \right. \\
& \quad \left. - \frac{b^{*2}}{2} C_1 + b^* C_1 l_s \dots - \frac{b^{*2}}{2} C_N + b^* C_N l_N \right] \\
& - \left[\frac{b^{*5} W}{30b^*} - \frac{b^{*3} C_{o,1}}{3} + \frac{b^{*2}}{2} C_{o,1} l_{o,1} \dots - \frac{b^{*3}}{3} C_{o,N} + \frac{b^{*2}}{2} C_{o,N} l_{o,N} \right. \\
& \quad \left. - \frac{b^{*3}}{3} R_s + \frac{b^{*2}}{2} R_s l_s - \frac{b^{*3}}{3} C_1 + \frac{b^{*2}}{2} C_1 l_s \dots - \frac{b^{*3}}{3} C_N + \frac{b^{*2}}{2} C_N l_N \right]
\end{aligned}$$

If the terms in Eq.(5-51) are written with sum symbols, Eq.(5-52) can be derived.

$$\begin{aligned}
\Delta_s = & -l_s \left[\frac{l_s^4 W}{24b^*} - \frac{l_s^2}{2} \sum_{k=1}^{N_o} C_{o,k} + l_s \sum_{k=1}^{N_o} C_{o,k} l_{o,k} - \frac{l_s^2}{2} R_s + l_s R_s l_s \right] \\
& + \left[\frac{l_s^5 W}{30b^*} - \frac{l_s^3}{3} \sum_{k=1}^{N_o} C_{o,k} + \frac{l_s^2}{2} \sum_{k=1}^{N_o} C_{o,k} l_{o,k} - \frac{l_s^3}{3} R_s + \frac{l_s^2}{2} R_s l_s \right] \\
& + l_s \left[\sum_{k=1}^{N_{in}} \left(\frac{l_k^2}{2} C_k - l_k C_k l_k \right) \right] - \left[\sum_{k=1}^{N_{in}} \left(\frac{l_k^3}{3} C_k - \frac{l_k^2}{2} C_k l_k \right) \right] \\
& + l_s \left[\frac{b^{*3} W}{24} - \frac{b^{*2}}{2} \sum_{k=1}^{N_o} C_{o,k} + b^* \sum_{k=1}^{N_o} l_{o,k} C_{o,k} - \frac{b^{*2}}{2} R_s + b^* R_s l_s - \frac{b^{*2}}{2} \sum_{k=1}^{N_{in}} C_k \right. \\
& \quad \left. + b^* \sum_{k=1}^{N_{in}} C_k l_k \right] \\
& - \left[\frac{b^{*5} W}{30b^*} - \frac{b^{*3}}{3} \sum_{k=1}^{N_o} C_{o,k} + \frac{b^{*2}}{2} \sum_{k=1}^{N_o} l_{o,k} C_{o,k} - \frac{b^{*3}}{3} R_s + \frac{b^{*2}}{2} R_s l_s \right. \\
& \quad \left. - \frac{b^{*3}}{3} \sum_{k=1}^{N_{in}} C_k + \frac{b^{*2}}{2} \sum_{k=1}^{N_{in}} C_k l_k \right]
\end{aligned} \tag{5-52}$$

If the following steps are applied to Eq.(5-52);

- the terms in Eq.(5-52) are distributed into parentheses.
- Assumption of $\Delta_s = 0$ is applied.
- The equation is rearranged to isolate R_s on the left side of the equation.
- And similar terms are collected to arrange the final equation better.

After applying these steps, Eq.(5-53) can be obtained finally. Using Eq.(5-53), the strut reaction force R_s can be calculated in the simplified indeterminate TBW structural model given in Figure 51 for unlimited numbers of concentrated loads positioned at the inner or the outer sections of the TBW.

$$\begin{aligned}
R_s = & \left[-\frac{l_s^5 W}{120b^*} + \frac{l_s b^{*3} W}{24} - \frac{b^{*4} W}{30} + \sum_{k=1}^{N_o} C_{o,k} \left(\frac{l_s^3}{6} - \frac{l_s b^{*2}}{2} + \frac{b^{*3}}{3} \right) \right. \\
& + \sum_{k=1}^{N_o} C_{o,k} l_{o,k} \left(-\frac{l_s^2}{2} + l_s b^* - \frac{b^{*2}}{2} \right) + l_s \sum_{k=1}^{N_{in}} \left(-\frac{l_k^2}{2} C_k \right) \\
& - \sum_{k=1}^{N_{in}} \left(-\frac{l_k^3}{6} C_k \right) + \sum_{k=1}^{N_{in}} C_k \left(-\frac{l_s b^{*2}}{2} + \frac{b^{*3}}{3} \right) \\
& \left. + \sum_{k=1}^{N_{in}} C_k l_k \left(l_s b^* - \frac{b^{*2}}{2} \right) \right] / \left[\frac{l_s^3}{3} + l_s b^{*2} - l_s^2 b^* - \frac{b^{*3}}{3} \right]
\end{aligned} \tag{5-53}$$

5.3.4 Validation of the Simplified Redundant SBW Models

The previous sections introduce two simplified indeterminate SBW models to compute the strut reaction force. In this section, the analytical solutions to computing the strut reaction forces in those models are validated by comparing their results to the result of the finite element analysis (FEA) run in SolidWorks.

Table 14 Input parameters used for the validation of the simplified models.

Description	Inputs	Value	Unit	Inputs for Model1	Inputs for Model2
ND spanwise position of the strut	Λ_{Strut}	0.50		l_1 , mm	l_s , mm
ND spanwise position of the engine	Λ_{Engine}	0.15		l_e , mm	l_2 , mm
Half span length	$b/2$	50,000.00	mm	l , mm	$b/2$, mm
Engine weight	W_E	1,000.00	N	C_{Engine} , N	C_2 , N
The magnitude of the distributed load at the root	W	0.04	N/mm	W , N/mm	W , N/mm

In this section, the simplified model presented in Section - 5.3.2 for one concentrated load is called Model-1, while the model with an unlimited number of concentrated loads in Section 5.3.3 is called Model-2. The two models have different symbols for the input variables. Table 14 presents the input parameters with the symbols of the models. The strut (Λ_{Strut}) and engine (Λ_{Engine}) locations are ND parameters as a fraction of the half span length hence input parameters of Model-1 and Model-2 can be computed regarding this.

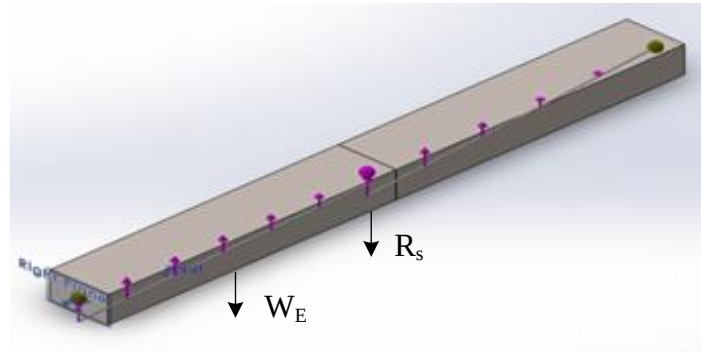


Figure 52 Beam model generated in SolidWorks for the validation study.

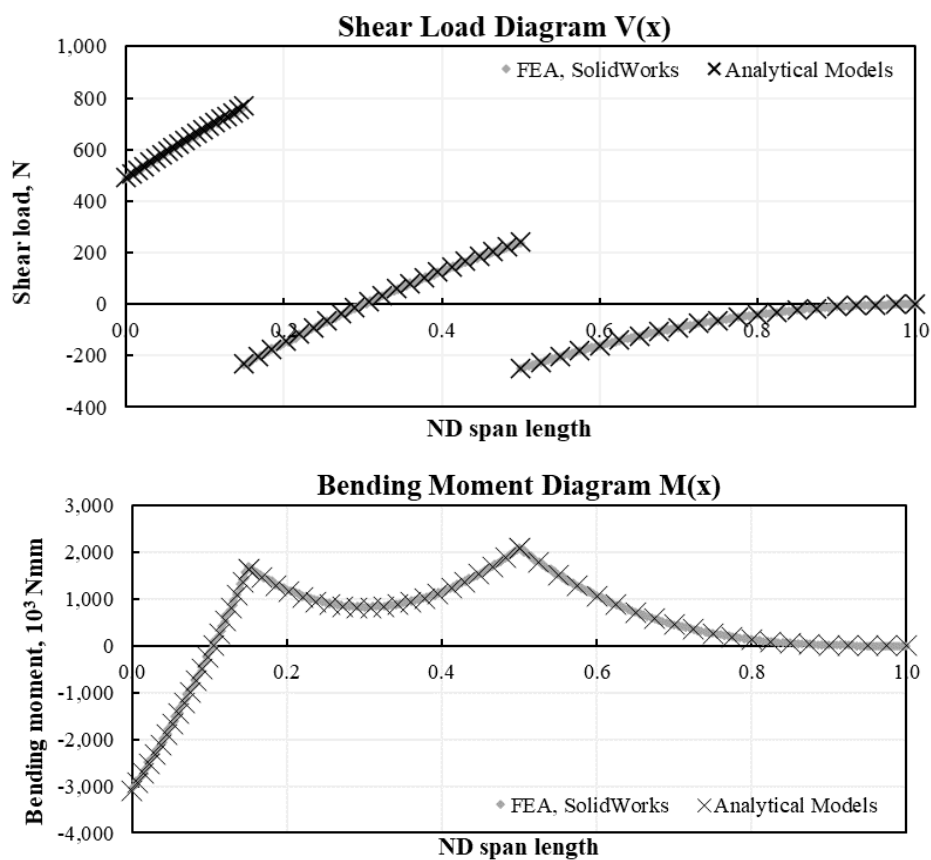


Figure 53 Shear force and bending moment diagrams generated for the analytical models and exported from SolidWorks.

The illustrations of Model-1 and Model-2 are presented in Figure 50 and Figure 51. The beam model generated in SolidWorks is depicted in Figure 52. In all the models, the beams are fixed at the left-hand sides, and the struts are pinned to the beams. The load distribution has a triangular shape between the root and the tip of the beam.

Table 15 Comparison of the strut reaction forces computed with the analytical models and FEA.

	Model-1, R_C	Model-2, R_S	FEA, R_S	Difference
Strut reaction force, N	491.00	491.00	489.00	-0.41%

The validation results can be seen in Table 15 and Figure 53. Both analytical methods computed the same strut reaction forces of 491(N) for the inputs in Table 14, while the FEA model computed the reaction force of 489(N). The difference between the strut reaction force calculation of the FEA and analytical models is -0.41%. The moment and shear diagrams are plotted in Figure 53 for the given input parameters. The shear load and bending moment values of the FEA model are extracted from SolidWorks. The analytical and FEA methods' shear and bending moment values are almost identical. Considering the validation results, it can be said that the analytical solutions for the simplified TBW models are accurate enough.

5.4 Structural Analysis of the Juries

The juries are supportive structural components in a TBW configuration and are used to increase the critical buckling load of the strut by changing its buckling modes. The juries are attached from their ends to the wing and the strut. In this section, an analytical method will be introduced for the sizing and ME of the juries.

The loads applied on this structural component should be known to size the juries. The primary axial load applied to the juries are caused by the strut when the strut tends to buckle under compressive loads. Hence a conservative structural model can be generated to simulate the buckling of the strut.

Two different TBW concepts are modelled. In one model, the juries are designed normal to the wing at one end and coincide with the strut at the other end with an angle. The other model has the juries normal to the strut and with an angle to the wing at the joint.

The load and displacement transformations are considered in 3D dimensions in both models. Hence all the geometric relations are covered in the derivations without any simplifications.

5.4.1 Strut Buckling Models

5.4.1.1 One Jury Model

If the strut is not supported, it will likely have the first buckling mode and bend, as shown in Figure 54. Due to the bending stiffness of the strut, the deflected strut will still carry the load until it exceeds the bifurcation point where the loads are in equilibrium and critical buckling load, P_{cr} , is reached (Hibbeler, 2015).

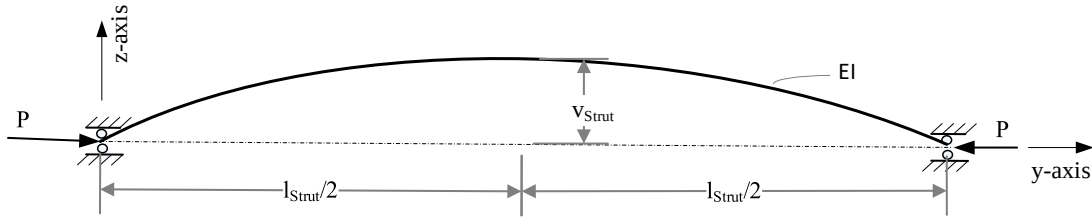


Figure 54 Strut column buckling the first mode.

The buckling mechanism of the strut in Figure 54 can be modelled as shown in Figure 55, where the spring with the stiffness of k_1 represents the bending stiffness of the strut. The force generated by the spring is $F_{Strut} = k_1 v_{Strut}$. The disturbing force is $F_{Disturbing}$ and it is caused by the compressive loads P .

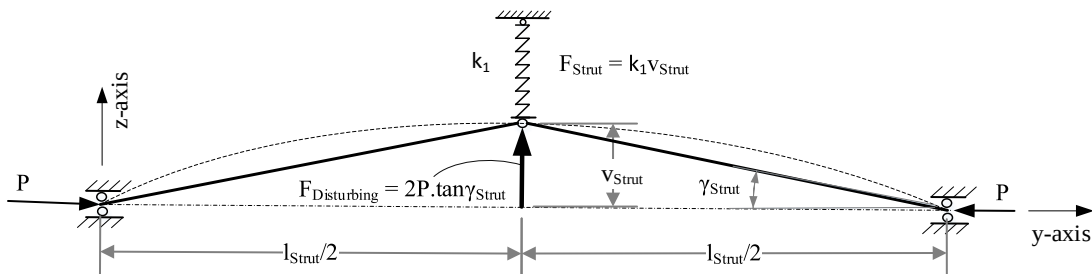


Figure 55 Representative buckling model for the first buckling mode of the strut.

If F_{Strut} is greater than $F_{Disturbing}$, this condition is called stable equilibrium and F_{Strut} will restore the strut to its unloaded position. If F_{Strut} is equal to $F_{Disturbing}$, the condition is called neutral equilibrium, and the load P is equal to P_{cr} which is also called a bifurcation point. Any increase in P after the bifurcation point will make the system unstable, and a sudden buckling will occur.

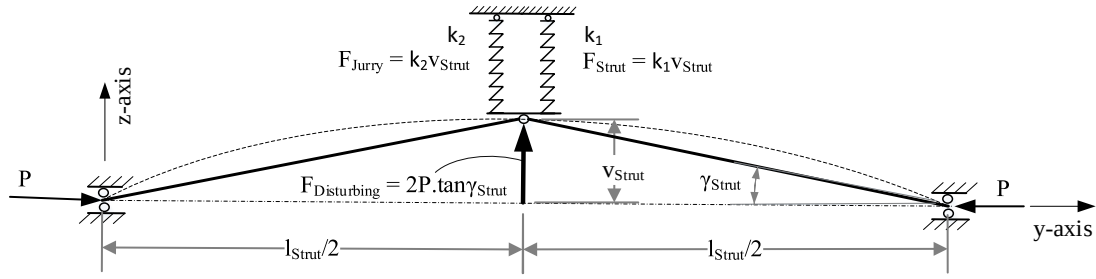


Figure 56 Simplified TBW model with jury support.

The jury's purpose is to prevent the strut from buckling at its attachment point by providing additional stiffness to the strut, as illustrated in Figure 56.

If we make a conservative assumption and accept that the stiffness of the strut, k_1 is zero; therefore, all the loads will be carried by the jury. Then the jury should be designed to resist the $F_{Disturbing}$ load without failure or exceeding its yield stress.

To be able to find $F_{Disturbing}$, v_{Strut} should be known. As can be seen in Figure 56, v_{Strut} will be equal to the axial deflection of the jury due to the compressive load, $F_{Disturbing}$. The greater v_{Strut} , the greater $F_{Disturbing}$ will be. Hence the highest value of v_{Strut} , can be calculated from the highest axial deflection of the jury, which will occur at its yield strain as the jury should be designed not to exceed ϵ_Y . After finding the v_{Strut} , $F_{Disturbing}$ can be calculated simply with geometric relations shown in Figure 56. Finally, the jury can be sized to resist $F_{Disturbing}$ without buckling or exceeding its yielding stress.

5.4.1.2 Other Modes

Similar to the assumption made in the previous section, the strut bending stiffness is accepted as zero to have a more conservative model. If the jury has an infinite axial stiffness, there will not be an axial deflection of the jury, and the strut is likely to buckle, as shown in Figure 57.

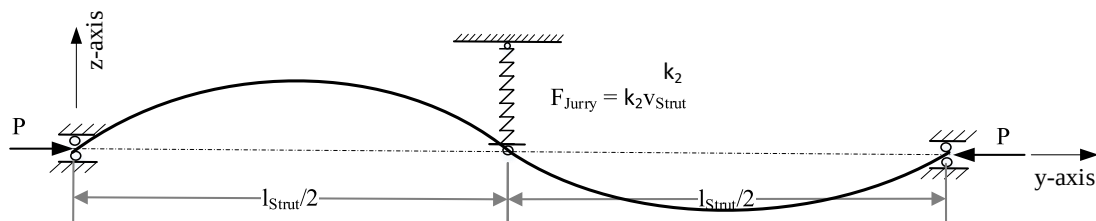


Figure 57 Buckling mode of the strut with infinite axial stiffness of the jury.

If the bending stiffness of the strut is assumed to be zero and the jury is deflected, as given in Figure 58, all the disturbing loads are assumed to be carried by the jury. The calculation process of $F_{Disturbing}$ is similar to the process explained in the previous section.

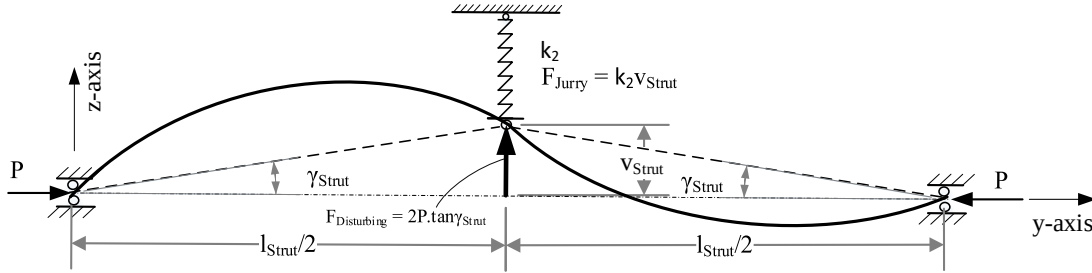


Figure 58 Buckling mode of the strut with lower axial stiffness of the jury.

5.4.1.3 Two Jury Model

The simplified TBW design with two juries is illustrated in Figure 59. It can be observed that the second jury attached at point B will contribute to resisting the disturbing forces caused by the compressive load P and reduce the axial load carried by the first jury attached at point A. The restoring effect of the second jury will be neglected to reduce the complexity and have a more conservative jury sizing model while sizing the first jury. And the same assumption will be applied in sizing the second jury. Hence the axial load applied to the first jury, $F_{Disturbing,1}$, will be equal to $F_{Disturbing} = P.tan\gamma_{Strut} + P.tan\phi_{Strut}$.

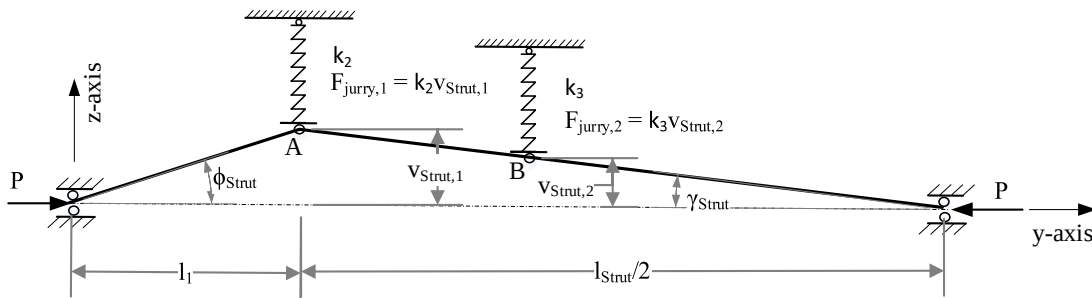


Figure 59 Strut design with two juries.

The 3D application of the simplified jury sizing models will be explained in the following sections.

5.4.2 Juries Normal to the Wing

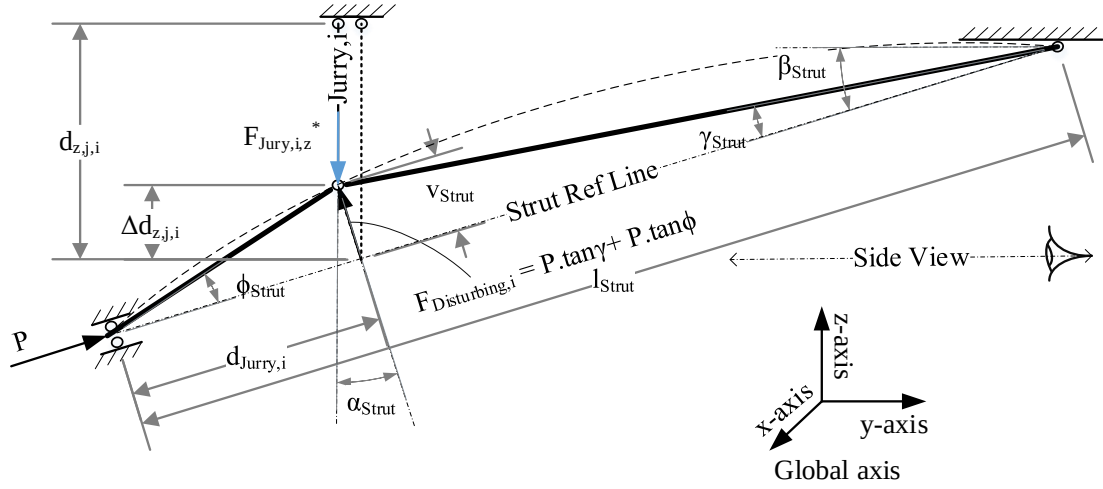


Figure 60 The front view of the structural model for sizing juries when designed vertically to the lower surface of the wing box.

The juries can be designed normal to the lower surface of the wing box. The calculation process of the axial load acting on the juries ($F_{Jury,i}$) is explained in this section. First $F_{Disturbing}$ can be formulated as in Eq.(5-54).

$$F_{Disturbing} = P.tan\phi_{Strut,i} + P.tan\gamma_{Strut,i} \quad (5-54)$$

where,

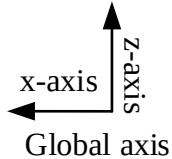
$$\begin{aligned} tan\phi_{Strut,i} &= \frac{v_{Strut,i}}{d_{jury,i}} \\ tan\gamma_{Strut,i} &= \frac{v_{Strut,i}}{l_{Strut} - d_{jury,i}} \end{aligned} \quad (5-55)$$

From the geometric relations illustrated in Figure 60, the formula in Eq.(5-56) can be written for $v_{Strut,i}$.

$$v_{Strut,i} = \frac{\Delta d_{z,Jury,i}}{cos\alpha_{Strut}} \quad (5-56)$$

In Eq.(5-56), $cos\alpha_{Strut}$ is equal to $cos\beta_{Strut}$ when the jury is designed normal to the wing, it can be calculated with the following formula in Eq.(5-57).

(5-57)



in Eq.(5-56), $\Delta d_{z,Jury,i}$ can be calculated with Eq. (5-58) using the geometric model in Figure 61.

(5-58)

$$l_{jury,i}^* = l_{jurry,i} - \Delta l_{jury,i} \quad (5-59)$$

(5-59)

$$\Delta l_{Jury,i} = \varepsilon_Y l_{Jurry,i} \quad (5-60)$$

(5-60)

$$A_1 = \sqrt{(1 - \frac{1}{2})^2 + 1^2} \quad (5-61)$$

(5-61)

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$$v_{Strut,i} = \frac{d_{z,Jury,i} - \sqrt{(l_{Jury,i} - \varepsilon_Y l_{Jury,i})^2 - d_{x,Jury,i}^2}}{\cos \beta_{Strut}} \quad (5-62)$$

v_{Strut} in Eq.(5-62) can be substituted in (5-55) and $\tan \phi_{Strut,i}$ and $\tan \gamma_{Strut,i}$ can be computed with Eq.(5-84).

$$\tan \phi_{Strut,i} = \frac{d_{z,Jury,i} - \sqrt{(l_{Jury,i} - \varepsilon_Y l_{Jury,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i} \cos \alpha_{Strut}} \quad (5-63)$$

$$\tan \gamma_{Strut,i} = \frac{d_{z,Jury,i} - \sqrt{(l_{Jury,i} - \varepsilon_Y l_{Jury,i})^2 - d_{x,Jury,i}^2}}{\cos \alpha_{Strut} (l_{Strut} - d_{Jury,i})}$$

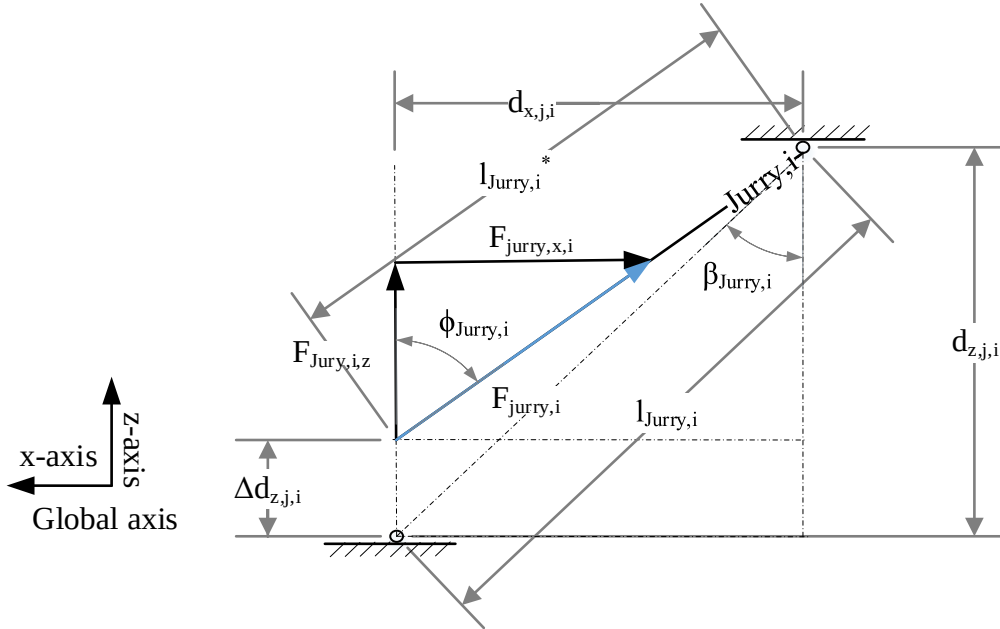


Figure 62 The side view (in the direction shown in Figure 60) of the structural model for sizing juries when they are designed vertically to the upper surface of the strut-box to calculate geometric relations between loads.

If the terms in Eq.(5-63) are substituted in Eq. (5-54), $F_{Disturbing,i}$ can be formulated as given in Eq. (5-64).

$$F_{Disturbing,i} = P \frac{d_{z,Jury,i} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i} \cos \alpha_{Strut}} + P \frac{d_{z,Jury,i} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{\cos \alpha_{Strut} (l_{Strut} - d_{Jury,i})} \quad (5-64)$$

Using the method of the joint at the strut-jury joint with the help of the illustration in Figure 60, $F_{Jury,i,z}$ can be calculated with Eq.(5-65).

$$F_{Jury,i,z} = \frac{F_{Disturbing}}{\cos \alpha_{Strut}} \quad (5-65)$$

After finding $F_{Disturbing,i,z}$, the resultant force acting on the jury, $F_{Jury,i}$ can be formulated as given in Eq.(5-66) with the help of Figure 62.

$$F_{Jury,i} = \frac{F_{Jury,i,z}}{\cos \phi_{Jury,i}} \quad (5-66)$$

$\cos \phi_{Jury,i}$ in Eq.(5-66) can be computed with the formula in Eq.(5-67).

$$\cos \phi_{Jury,i} = \frac{d_{z,Jury,i} - \Delta d_{z,Jury,i}}{l_{Jurry,i}^*} \quad (5-67)$$

In Eq.(5-67), $l_{Jurry,i}^*$ is given in Eq. (5-59) and $\Delta d_{z,Jury,i}$ is given in Eq.(5-61). Hence $\cos \phi_{Jury,i}$ can be written as given in Eq.(5-68).

$$\cos \phi_{Jury,i} = \frac{\sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{l_{Jurry,i} - \varepsilon_Y l_{Jurry,i}} \quad (5-68)$$

If Eqs.(5-68)-(5-65) are substituted in Eq.(5-66), finally $F_{Jury,i}$ can be computed with the formula given in Eq.(5-69).

$$F_{Jury,i} = \left(P \frac{d_{z,Jury,i} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i} \cos \alpha_{Strut}} \right. \\ \left. + P \frac{d_{z,Jury,i} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{\cos \alpha_{Strut} (l_{Strut} - d_{Jury,i})} \right) \\ * \frac{1}{\cos \alpha_{Strut} * \cos \phi_{Jury,i}} \quad (5-69)$$

5.4.3 Juries Normal to the Strut

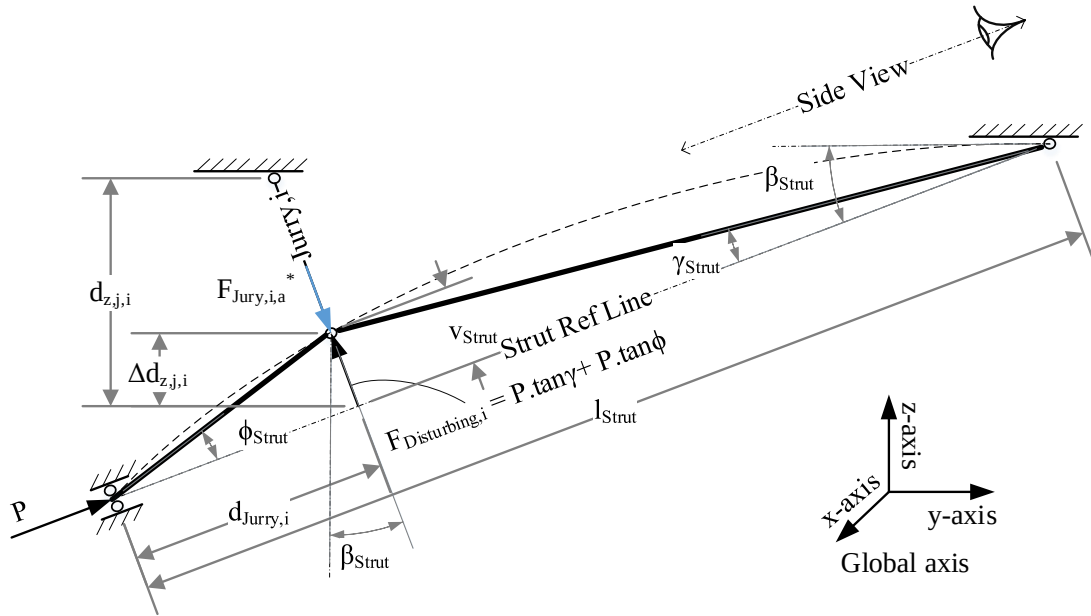


Figure 63 The front view of the structural model for sizing juries when designed normal to the upper surface of the strut box.

The sizing procedure of the juries positioned normal to the upper surface of the strut box is explained in this section. The axial disturbing load, resisted by the jury, can be calculated with Eq.(5-70).

$$F_{Disturbing,i} = P.tan \phi_{Strut,i} + P.tan \gamma_{Strut,i} \quad (5-70)$$

In Eq.(5-70), $tan \gamma_{Strut,i}$ and $tan \phi_{Strut,i}$ can be calculated as given in (5-71).

$$tan \phi_{Strut,i} = \frac{v_{Strut,i}}{d_{Jury,i}}$$

$$\tan \gamma_{Strut,i} = \frac{v_{Strut,i}}{l_{Strut} - d_{Jury,i}} \quad (5-71)$$

In Eq.(5-71), $v_{Strut,i}$ is unknown and a formula can be derived for its computation as given in Eq.(5-72) using the Pythagorean theorem with the triangular geometry given in Figure 64. It should be noted that the components are located on the jury axis in Figure 64, and their dimensions are transformed from the global axis to the jury axis.

$$v_{Strut,i} = \frac{d_{z,Jury,i}}{\cos \beta_{Strut}} - \sqrt{l_{Jury,i}^{*2} - d_{x,Jury,i}^2} \quad (5-72)$$

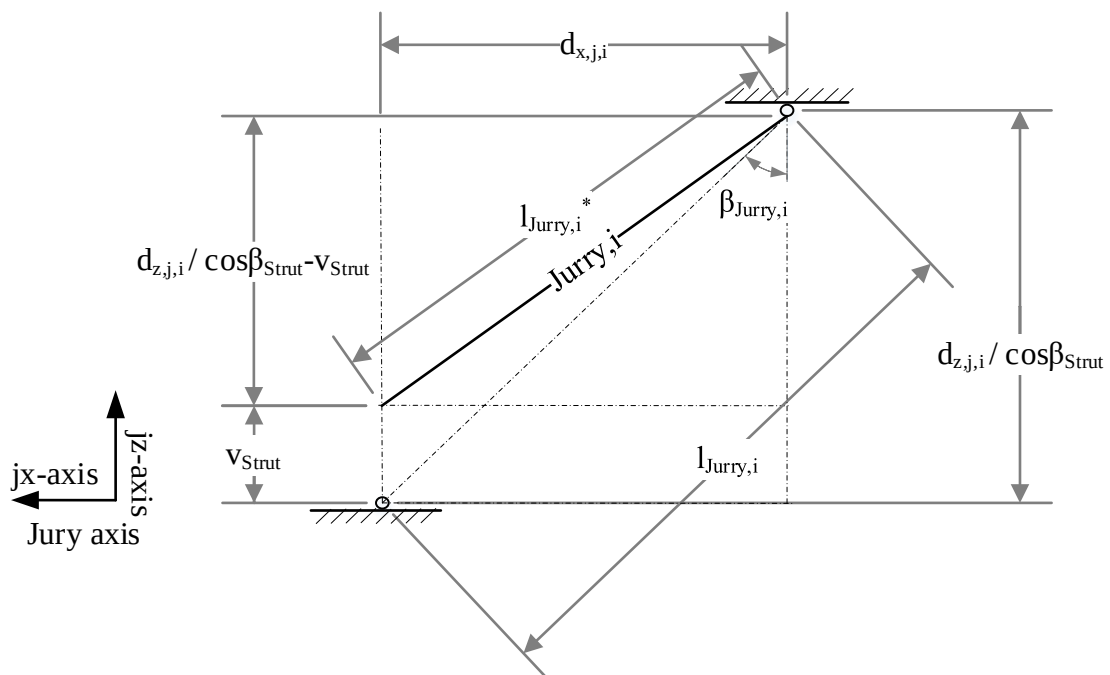


Figure 64 The side view (in the direction shown in Figure 63) of the structural model for sizing juries when they are designed vertically to the upper surface of the strut-box to calculate geometric relations.

In Eq.(5-72), $l_{jury,i}^*$ and $\cos\beta_{Strut}$ should be known. $\cos\beta_{Strut}$ can be calculated using Eq. (5-57) and $l_{jury,i}^*$ can be calculated from ε_Y as given in Eq.(5-73), if the jury is sized for yielding stress.

$$l_{Jury,i}^* = l_{Jurry,i} - \Delta l_{Jurry,i} \quad (5-73)$$

$$\Delta l_{lurry,i} = \varepsilon_Y l_{lurry,i} \quad (5-74)$$

If Eqs. (5-73) and (5-74) are substituted in Eq.(5-72), v_{Strut} can be computed with Eq.(5-75).

$$v_{Strut} = \frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2} \quad (5-75)$$

If v_{Strut} from Eq.(5-75) is substituted in Eq.(5-71), $\tan\phi_{Strut}$ and $\tan\gamma_{Strut}$ can be found using Eq.(5-76).

$$\tan\phi_{Strut} = \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i}} \quad (5-76)$$

$$\tan\gamma_{Strut} = \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{(l_{Strut} - d_{Jury,i})}$$

If terms in Eq. (5-76) are substituted in Eq.(5-70), Eq.(5-77) can be derived to find the disturbing load, $F_{Disturbing}$.

$$F_{Disturbing,i} = P \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i}} \quad (5-77)$$

$$+ P \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{(l_{Strut} - d_{Jury,i})}$$

It can be seen that if the Method of Joint is applied to the truss system in Figure 63, $F_{Jury,a,i}^*$ is equal but in an opposite direction to $F_{Disturbing,i}$. $F_{Jury,a,i}$ is the component of the axial compressive resultant load ($F_{Jury,i}$) acting on the jury. Hence, the resultant load $F_{Jury,i}$ can be computed using Eq. (5-78) using the geometric relations in Figure 65.

$$F_{Jury,i} = \frac{F_{Jury,a,i}}{\cos\phi_{Jury,i}} \quad (5-78)$$

The term $\cos\phi_{Jury,i}$ in Eq.(5-78) can be computed as given in Eq.(5-91).

$$\cos\phi_{Jury,i} = \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - v_{Strut}}{l_{Jurry,i}^*} \quad (5-79)$$

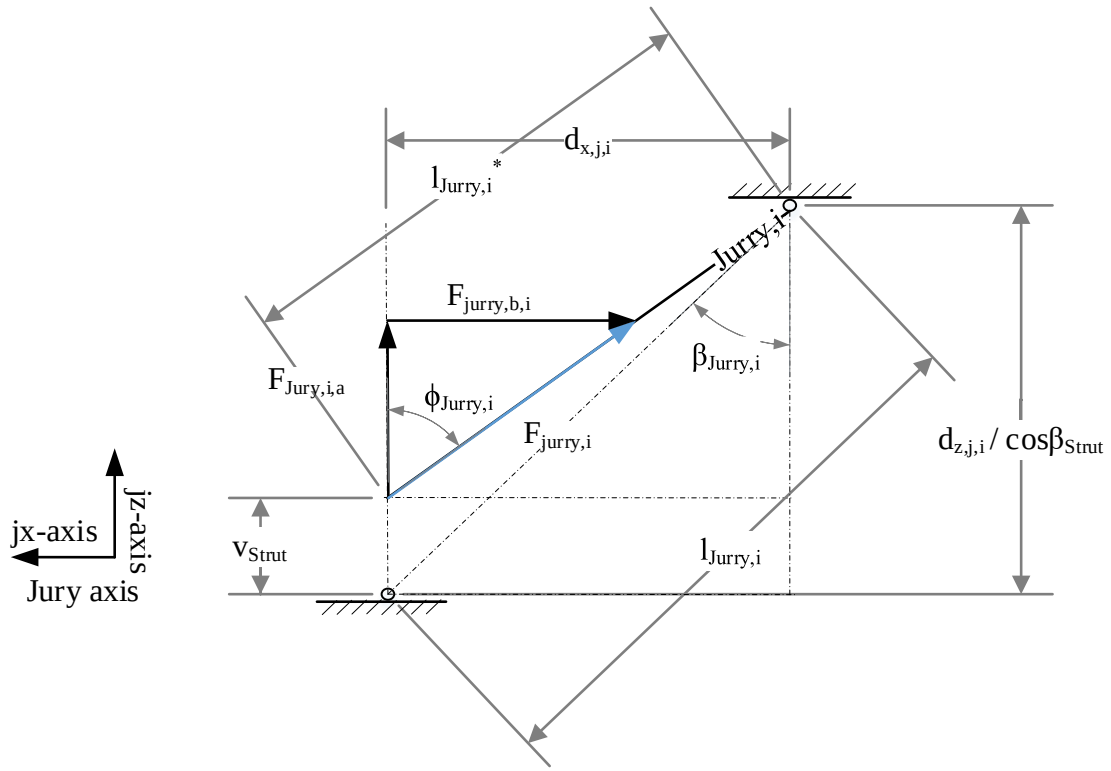


Figure 65 The side view (in the direction shown in Figure 63) of the structural model for sizing juries when they are designed normal to the upper surface of the strut-box to calculate geometric relations between loads.

In Eq.(5-79), $l_{Jury,i}^*$ and v_{Strut} are given in Eq.(5-73) and (5-75), respectively. If they are substituted in Eq.(5-79), Eq.(5-80) can be derived for $\cos \phi_{Jury,i}$.

$$\cos \phi_{Jury,i} = \frac{\sqrt{(l_{Jury,i} - \varepsilon_Y l_{Jury,i}^*)^2 - d_{x,Jury,i}^2}}{l_{Jury,i} - \varepsilon_Y l_{Jury,i}} \quad (5-80)$$

If Eq.(5-80) and Eq.(5-77) are substituted in Eq.(5-78), finally $F_{Jury,i}$ can be computed with the final equation given in Eq. (5-81).

$$F_{Jury,i} = P \left(\frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{d_{Jury,i}} \right. \quad (5-81)$$

$$\left. + P \frac{\frac{d_{z,Jury,i}}{\cos\beta_{Strut}} - \sqrt{(l_{Jurry,i} - \varepsilon_Y l_{Jurry,i})^2 - d_{x,Jury,i}^2}}{(l_{Strut} - d_{Jury,i})} \right)$$

$$* \frac{1}{\cos\phi_{Jury,i}}$$

After computing the axial loads applied to the juries using Eqs (5-81) or (5-69) for the juries normal to the strut or the wing, respectively, the jury can be sized for yield strength and column buckling using the methods explained in Section 5.5.1 Sizing of Rectangular Box. If the TBW concept has two juries, each jury can be sized using the same conservative procedure explained in Section 5.4.1.3 Two Jury Model.

5.5 Stress Analyses and Sizing of the Truss Components

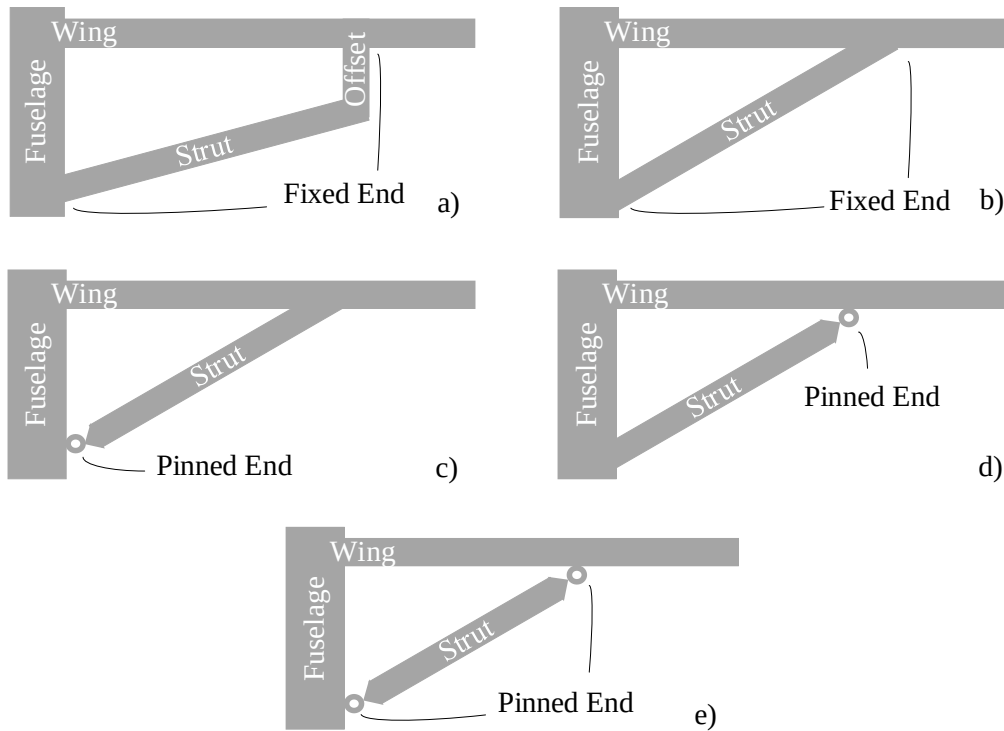


Figure 66 Boundary conditions of the SBW frame with and without offset.

SBW structure is a frame comprising the strut, offset, juries, wing, and fuselage. Possible five frames for SBW configuration with different boundary conditions are shown with simplified illustrations in Figure 66. The frame can be designed with and without an offset. In this section, sizing methods for these components will be developed considering the boundary conditions at their joints.

5.5.1 Sizing of Rectangular Box

A rectangular box model for the load-carrying structures of the strut, jury and offset is introduced In Chapter 3. Its sizing procedure is explained in this section.

Different loading cases are considered in the sizing of the strut, offset and juries. Their load-carrying structures are sized to resist the column and local buckling stresses due to the bending and axial loads.

5.5.1.1 Euler Column Buckling

In this section, the formula to calculate the minimum thickness of the rectangular box to resist the column buckling is derived. The well-known equation of the Euler column buckling for the n^{th} number of the buckling modes is given in Eq. (5-82)

$$P_{Cr,n} = \frac{n^2 \pi^2 EI}{l_{Ef,Comp}^{*2}} \quad (5-82)$$

Where $l_{Ef,Comp}^*$ is the effective length of the column considering the boundary conditions and the juries, n is the number of the buckling mode. From the inspection, it can be noticed that buckling will occur around the x_{local} the axis of the rectangular box given in Figure 20. And Eq. (5-82) can be written for the first buckling mode of the column with a rectangular box as in Eq. (5-83).

$$P_{Cr,1} = \frac{\pi^2 EI_{xx}}{l_{Ef,Comp}^{*2}} \quad (5-83)$$

In this equation, the values of all the parameters are known except the moment of inertia of the rectangular box, I_{xx} . The I_{xx} of the rectangular box is given in Eq. (5-84).

$$I_{xx} = \frac{w_{Comp,box,j} * t_{Comp,Skin,Bk,j} * t_{Comp,box,j}^2}{2} + \frac{2t_{Comp,Skin,Bk,j} * t_{Comp,box,j}^3}{12} \quad (5-84)$$

which can be reorganised as in Eq. (5-85).

$$I_{xx} = \frac{t_{Comp,Skin,Bk,j}(3w_{Comp,box,j} * t_{Comp,box,j}^2 + t_{Comp,box,j}^3)}{6} \quad (5-85)$$

If Eq. (5-85) is substituted in Eq. (5-83), the term $t_{Strut,Skin,Bk,j}$ can be isolated as given in Eq. (5-86).

$$t_{Comp,Skin,Bk,j} = \frac{6P_{Cr,1}l_{Ef,Comp}^2}{\pi^2 E(3w_{Comp,box,j} * t_{Comp,box,j}^2 + t_{Comp,box,j}^3)} \quad (5-86)$$

The term $t_{Comp,Skin,Bk,j}$ is the required skin thickness to resist the column buckling and can be calculated with Eq. (5-86). In Eq. (5-86) l_{Comp}^* is the length of the component. For the strut, the length of the component will be equal to the strut section length (l_{Strut}^*) as illustrated in Figure 67.

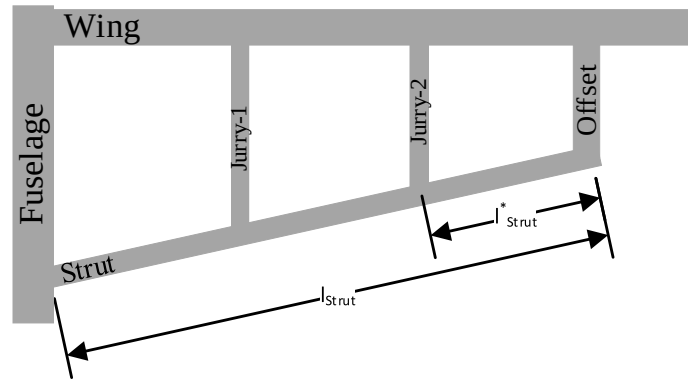


Figure 67 Illustration of the strut sections and their length.

The different boundary condition combinations of SBW configuration are presented in Figure 66. After calculating the l_{Comp}^* , $l_{Ef,Comp}^*$ can be calculated for different boundary conditions using Table 16 from Ref. (Megson, 2013).

Table 16 Colum effective length for different boundary conditions.

Column Ends	$l_{Ef,Comp}^*/l_{Comp}^*$
Both pinned	1.0
Both fixed	0.5
One fixed, the other pinned	0.6998

5.5.1.2 Strength at Yield or Ultimate Stress

The rectangular box is also sized not to exceed the yield or ultimate stress when the tensile or compression loads act on it. The stress acting on the box due to the compression or the tensile loads can be calculated by Eq. (5-87).

$$\sigma = \frac{P}{2(w_{Comp,box,j} * t_{Comp,skin} + t_{Comp,box,j} * t_{Comp,skin})} \quad (5-87)$$

If Eq. (5-87) is rearranged for yield stress and the $t_{Strut,skin}$ is isolated, the thickness to resist the axial loads, $t_{Strut,skin,St}$ can be found with Eq. (5-88).

$$t_{Comp,skin,St,j} = \frac{P}{2\sigma_Y(w_{Comp,box,j} + t_{Comp,box,j})} \quad (5-88)$$

As previously explained in detail, σ_Y is used for isotropic materials under limited loads while σ_U is used for composite materials under the ultimate loads.

5.5.1.3 Bending Resistance

Bending stresses are expected to dominate in the offset and jury sizing. The maximum axial stress on the rectangular box due to the bending moment and the tensile or compression stresses can be calculated by Eq. (5-89).

$$\sigma_{Max} = -\frac{M_x y_o}{I_{xx}} + \frac{M_y x_o}{I_{yy}} + \frac{P}{A_{Comp,box,j}} \quad (5-89)$$

In the previous section I_{xx} , was calculated as given in Eq. (5-90).

$$I_{xx} = \frac{t_{Comp,skin,Bdj}(3w_{Comp,box,j} * t_{Comp,box,j}^2 + t_{Comp,box,j}^3)}{6} \quad (5-90)$$

Similarly, I_{yy} can be calculated using Eq. (5-91).

$$I_{yy} = \frac{t_{Comp,skin,Bdj}(3t_{Comp,box,j} * w_{Comp,box,j}^2 + w_{Comp,box,j}^3)}{6} \quad (5-91)$$

The maximum stresses occur close to the surface of the skin of the rectangular boxes hence, x_o and y_o can be calculated by Eq. (5-92).

$$y_o = \frac{t_{Comp,box,j}}{2}, \quad x_o = \frac{w_{Comp,box,j}}{2}, \quad (5-92)$$

In Eq. (5-89) A_{wb} is the surface area of the box cross-section, and it can be calculated by Eq. (5-93).

$$A_{Comp,box,j} = 2t_{Comp,Skin,Bd}(w_{Comp,box,j} + t_{Comp,box,j}) \quad (5-93)$$

If I_{xx} , I_{yy} and A_{wb} from Eqs. (5-90), (5-91) and (5-93), substitute in (5-89) and the equation is rearranged for yield stress (σ_Y), $t_{Comp,Skin,Bd}$ can be isolated for bending resistance material calculations as given in Eq. (5-94). The yield stress can be changed with the ultimate stress under ultimate loads when composite material is chosen for the structure.

$$t_{Comp,Skin,Bd,j} = \left| \left(+ \frac{6M_x y_o}{(3w_{Comp,box,j} * t_{Comp,box,j}^2 + t_{Comp,box,j}^3)} - \frac{6M_y x_o}{(3t_{Comp,box,j} * w_{Comp,box,j}^2 + w_{Comp,box,j}^3)} + \frac{P}{2(w_{Comp,box,j} + t_{Comp,box,j})} \right) \frac{1}{\sigma_Y} \right| \quad (5-94)$$

5.5.1.4 Local Buckling

The strut under compression is prone to local buckling. This section will introduce a simple direct sizing process considering the local buckling constraint of a strut with a rectangular box model.

A simple, critical local buckling stress calculation process is provided by (Mr M.E. Grayley, 1988). The method in the reference will be compiled to be integrated into the methods presented in this thesis. (Mr M.E. Grayley, 1988) assumes that the strut length is at least four times longer than the strut-box width or thickness.

From (Mr M.E. Grayley, 1988), the local buckling stress of a rectangular section tubular strut can be calculated with Eq. (5-95).

$$\sigma_{LocalBuck} = KE_{Material} \left(\frac{t_{Comp,Skin}}{w_{Comp,Box}} \right)^2 \quad (5-95)$$

Equation (5-95) gives the stress value above the local buckling starts. $\sigma_{LocalBuck}$ can also be calculated with Eq.(5-96).

$$\sigma_{LocalBuck} = \left(\frac{P_{Axial}}{A_{Comp,Box}} \right) \quad (5-96)$$

For the rectangular cross-section load-carrying box shown in Figure 20, the cross-sectional area of the box is given in Eq.(5-93). Hence Eq.(5-96) can be written as given

in Eq.(5-97). It should be noted that except $t_{Comp, Skin}$ in Eqs.(5-95) and (5-97), rest the other terms are known.

$$\sigma_{LocalBuck} = \left(\frac{P_{Axial}}{2t_{Comp, Skin, Bd}(w_{Comp, box, j} + t_{Comp, box, j})} \right) \quad (5-97)$$

If Eq.(5-97) is substituted in Eq.(5-95), and $t_{Comp, Skin, LB}$ are isolated Eq.(5-98) can be derived.

$$t_{Comp, Skin, LB} = \sqrt[3]{\frac{P_{Axial} w_{Comp, Box}}{2K_{Local} E_{Comp}(w_{Comp, box, j} + t_{Comp, box, j})}} \quad (5-98)$$

The term, K_{Local} , can be obtained from Figure 2 in Ref. (Mr M.E. Grayley, 1988). To find K_{Local} value, the ratio of the skin thickness of the short side of the rectangular box to that of the long side is required. For the model introduced in Section - 3.2.5 Rectangular Box Model, the skin thicknesses of both sides of the box are equal, and the ratio can be 1. And K value can be calculated from the relevant curve together with $t_{Comp, box, j} / w_{Comp, box, j}$. For quick, direct computation of K_{Local} , 5th order polynomial curve fitting is used, and the formula is given in Eq.(5-99). The polynomial in Eq. (5-99) has a R^2 value of 0.9997.

$$\begin{aligned} K_{Local} = & -18.702 \left(\frac{t_{Comp, box, j}}{w_{Comp, box, j}} \right)^5 + 53.638 \left(\frac{t_{Comp, box, j}}{w_{Comp, box, j}} \right)^4 \\ & - 61.223 \left(\frac{t_{Comp, box, j}}{w_{Comp, box, j}} \right)^3 + 34.015 \left(\frac{t_{Comp, box, j}}{w_{Comp, box, j}} \right)^2 \\ & - 10.41 \left(\frac{t_{Comp, box, j}}{w_{Comp, box, j}} \right) + 6.245 \end{aligned} \quad (5-99)$$

Hence, using Eq.(5-98), a direct calculation can be achieved to find the minimum skin thickness to resist the local buckling of the struts with the rectangular load-carrying box introduced in the previous sections.

5.5.1.5 Application of the Sizing Criteria

The sizing criteria used for each truss component, including strut, offset, jury-1, and jury-2, are listed in Table 17. The strut is sized for column buckling, strength and local buckling. For the computation of these sizing criteria, the axial load acting on the strut should be known, and the method for its computation is explained in 5.3 Solutions to the Indeterminate SBW Problem. In Section 5.3, the strut reaction force R_S is computed in

global z-axis. It can be seen in the free-body diagram in Figure 68 that the computed R_S should be transformed due to the load path changes. The loads can be transformed using the transformation matrix given in Section - 4.8.4 (Vector Transformations) to compute the resultant axial force ($P_{Axial,Strut}$ or F_{DC}) acting along the strut in the spanwise direction in Figure 68. The required geometric dimensions will be obtained from the geometric module explained in Section 3.3 Strut- and Truss-Braced Wings Geometric Model. After computing $P_{Axial,Strut}$, the strut box at the relevant strut section can be sized for column buckling, strength and local buckling using the methods explained in Section 5.5.1 Sizing of Rectangular Box.

Table 17 Sizing criteria used for each truss component, such as strut, offset, jury-1, and jury-2.

Sizing Criteria	Components Sized
Column Buckling	Strut, Offset, Jury-1, Jury-2
Strength at Yield or Ultimate Stress	Strut, Offset, Jury-1, Jury-2
Bending Resistance	Offset, Jury-1, Jury-2
Local Buckling of the Tube	Strut, Offset, Jury-1, Jury-2

To compute the axial load acting on the offset ($P_{Axial,Offset}$ or F_{BC} in Figure 68) similar procedure to that of the strut can be applied. The force F_{DC} shown in Figure 68 has two components of $F_{DC,z}$ and $F_{DC,y}$ in global z and y axes. They can be computed using vector transformations, as explained before. $F_{DC,y}$ will cause bending on the offset and this value will be the highest at the offset root. Section - 5.5.1.3 (Bending Resistance) explains sizing for bending resistance. In most sizing cases, this bending load is the most critical criterion. This criterion will be discussed in Section - 7.4.2 (Offset Design Parameters). When, $P_{Axial,Strut}$ or $F_{DC,z}$ is known, the offset box can be sized for column buckling, strength and local buckling using the methods explained in Section - 5.5.1 Sizing of Rectangular Box.

It should be noted that the free body diagram in Figure 68 is a simplified 2D illustration. As the geometric model and vector transformation matrix are provided for 3D dimension, the same procedure can be applied to the three-dimensional designs of the strut and offset. The methods presented in this study utilise vector transformations in 3D. As the 3D load computation of the juries requires extra treatments, they are already explained separately in Section 5.4 Structural Analysis of the Juries using 3D axes.

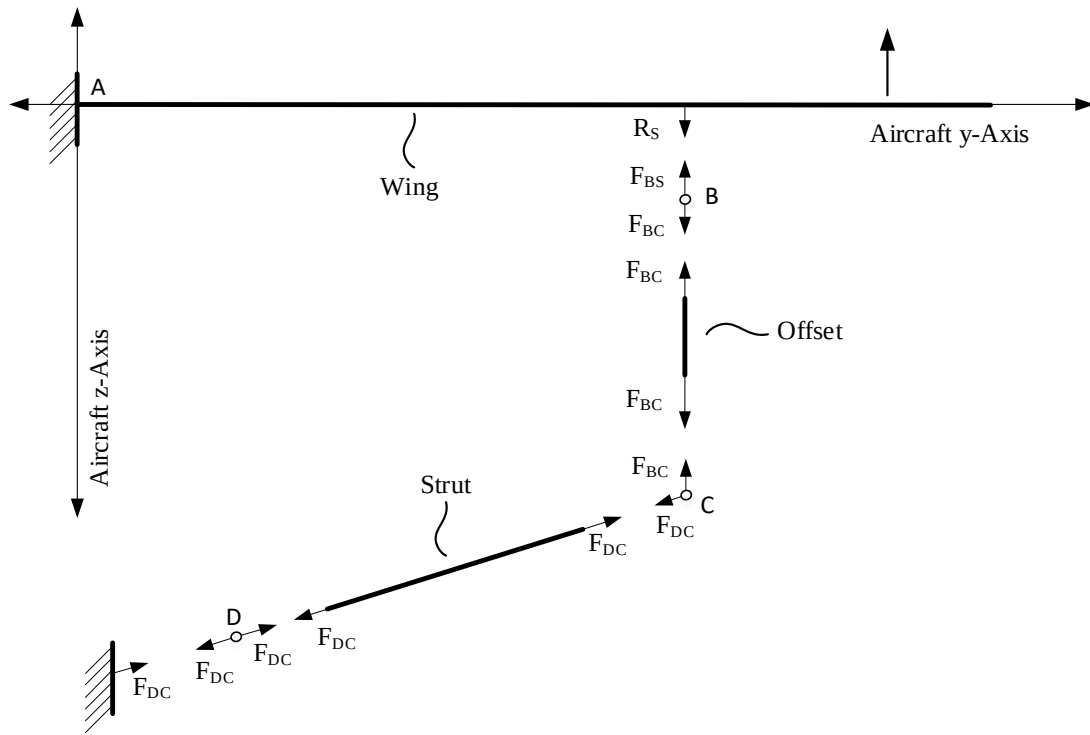


Figure 68 Free-body diagram of the strut, offset system.

The juries are sized for column buckling, strength, bending resistance and local buckling. To conduct the sizing procedure for column buckling, strength and local buckling; axial loads acting on the juries should be computed first. Detailed analytical methods are developed to compute those axial loads and explained in Section 5.4 Structural Analysis of the Juries.

In addition to the axial structural loads acting on the juries, the juries are also sized to resist the aerodynamic loads which cause bending on the juries. The simple conservative method is explained in Section - 4.6 (Aerodynamic Loads on Truss Components) for the computation of the aerodynamic loads. After finding the aerodynamic loads, the minimum skin thickness of the jury box can be computed using the procedure explained in Section 5.5.1.3 Bending Resistance.

5.5.1.6 Rectangular Box Skin Thickness

The procedures to find the minimum skin thickness of the rectangular box model are explained above following different constraints. Following the procedure, four thickness values will be calculated. The thicknesses, $t_{Comp,Skin,Bk,j}$, $t_{Comp,Skin,St,j}$, $t_{Comp,Skin,Bd}$, $t_{Comp,Skin,LB}$ denotes the minimum skin thickness required for the Euler column buckling, yield strength, bending resistance and local buckling constraints, respectively. The

selected thickness of the rectangular box should be the maximum of those values for a single loading case of n as formulated in Eq. (5-100).

$$t_{Comp,Skin,j,n} = \max(t_{Comp,Skin,Bk,j}, t_{Comp,Skin,St,j}, t_{Comp,Skin,Bd}, t_{Comp,Skin,LB}) \quad (5-100)$$

In Eq. (5-100), $t_{Comp,Skin,j,i,n}$ is the selected thickness for the component in n th loading case. All loading cases should be considered, and the computed thickness sizes should be recorded. The final skin thickness of the rectangular box should be the maxima in the list of skin thicknesses in Eq. (5-101)

$$t_{Comp,Skin,Selected,j} = \max(\{t_{Panel,j,i,n} : n = 1, \dots, N_{Case}\}) \quad (5-101)$$

5.5.2 Fixed Leading and Trailing Edges Sizing

In the previous section, the sizing procedure is explained for the skin of the load-carrying rectangular box. This section will explain the sizing method for the leading and trailing edges of the strut, jury and offset.

For a conventional aircraft, the LE of the wing should be sized to resist bird strikes, as explained in (Howe, 2004; EASA, 2020). According to (EASA, 2020), the aircraft should be able to continue to its safe flight after an impact with a bird of 1.81kg while flying at a speed of V_C at sea level or $0.85V_C$ at 2438m.

The procedure explained in (Howe, 2004) is used in this study to compute the thicknesses of the LE of truss components, including strut, offset and jury, as given in Eq. (5-102).

$$t_{Comp,LE,Skin,BS,j} = 0.02V_M m^{\frac{1}{3}} f(Mat) \cos^{\frac{2}{3}} \theta / [\exp(1230/(r^2 + 30r + 1000))] \quad (5-102)$$

Where, $t_{Comp,LE,Skin,BS}$ is the thickness of the LE of the component (mm), r is the leading edge nose radius (mm), m is the mass of the bird (kg), V_M is the impact velocity, θ is the inclination of the bird's strike direction normal to the surface. $f(Mat)$ represents a material factor, defined as $0.8[f_{-1}(Mat)/f_{-1}(L73)]$, where $f_{-1}(Mat)$ denotes the 0.1% proof stress of the selected material, and $f_{-1}(L73)$ refers to the 0.1% proof stress of the light alloy material, specified as L73.

Considering the simplicity of the calculation process, the same procedure is also employed to size the fixed TE thickness ($t_{Comp,TE,Skin,BS,j}$) of the truss components, too,

with more representative values of θ . It is taken 0 degrees for the LE and 30 degrees for the TE sizing.

5.6 Structural Material

In this study, the materials of the aircraft structures are grouped as isotropic and anisotropic. Isotropic materials are metals with consistent mechanical properties in all directions, while anisotropic materials, such as composites, exhibit different mechanical properties in different directions due to their inherent material structure. The use of these two types of material in the new model is explained in this section.

As seen in the previous sections, material properties are required in the structural sizing. And the density of the materials will also be used in the ME formulations in the following chapter. As discussed in Section - 4.1.2 Structural Sizing Criteria, the isotropic materials are sized for yield stress under limit loads. In contrast, the composite materials are sized with ultimate stress under the ultimate loads. Their application is discussed in the following sections.

5.6.1 Isotropic Materials

Their material properties are required to use the isotropic materials in the structural sizing and ME (SSaME) of the wing components. In the new MM, users can assign any isotropic materials. For the validation of the new method and other research, the isotropic materials of aluminium 7075 T6, aluminium 7075-T651, and aluminium 7178-T651 from ref. (York and Labell, 1980) were used and their properties are given in Table 18.

Table 18 The material properties of the isotropic materials used in this study.

Material Name	Density, kg/m^3	Tensile Modules, N/m^2	Shear Modulus, N/m^2
Al. 7075 T6	2,795.67	71,400,000,000.00	29,000,000,000.00
Al. 7075-T651	2,795.67	73,084,456,000.00	26,889,504,819.43
Al. 7178-T651	2,823.35	73,084,456,000.00	26,889,564,000.00
Material Name	Yield Strength, N/m^2	Ultimate Strength, N/m^2	Shear Strength, N/m^2
Al. 7075 T6	434,000,000.00	538,000,000.00	322,800,000.00
Al. 7075-T651	475,738,440.00	524,001,760.00	310,264,200.00
Al. 7178-T651	503,317,480.00	572,265,080.00	296,474,680.00

5.6.2 Anisotropic Materials

In this study, both metals and composites are considered as potential structural materials. Composites, being anisotropic materials, exhibit different stiffness values in

various directions. Consequently, they possess distinct elastic modulus values (E_x , E_y , and E_z), shear modulus values (G_{xy} , G_{yz} , and G_{xz}), and multiple Poisson's ratios in different directions. The design and analysis of composite materials can be computationally demanding, particularly during the initial design stage.

Although the anisotropic nature of composites allows for the design and optimization of materials according to applied loads, their design process can significantly increase the computational time when integrated into the MM developed in this thesis. As a solution, initial design studies can be conducted with quasi-isotropic composites to reduce computational time.

Composite materials can be optimized for thickness or weight objectives by altering their design variables through micromechanics or macro mechanics. If commercial laminas are chosen for composite implementation, their macro mechanics features can be optimized using classical laminate theory (CLT).

In this work, a computer program utilizing CLT has been developed to design and optimize composite materials. The module employs an optimizer that converges to a minimum composite thickness by adjusting the lamina orientation and their percentage in the laminate, given the knowledge of applied loads and their directions. Maximum stress failure criteria are used to prevent material failure in longitudinal and transverse directions. The module has been validated using publicly available composite design tools.

Despite the successful development and validation of the CLT-based module, it was observed that its implementation in the MM led to a considerable increase in computational time. This increase is attributed to the requirement of running the composite analysis and optimization tool for each wing box panel and mass boom in a wing section under every loading case.

To address this issue, a simplified approach using quasi-isotropic composites was employed for the initial design and sizing of composites, despite the availability of the CLT tool. This decision was made because the MM was developed for the conceptual and preliminary design stage of SDaME of novel aircraft wings, where computational time is critical. An advanced quasi-isotropic composite material from (Bradley, Allen and Droney, 2015) was employed for the design and validation studies. The material properties of the quasi-isotropic material are given in Table 19.

Table 19 Material properties of advanced quasi-isotropic composite (Bradley, Allen and Droney, 2015).

Material Name	Density, kg/m^3	Tensile Modules, N/m^2	Shear Modulus, N/m^2
Advanced Quasi-Isotropic Composite	1,577.8	72,394,820,667.7	26,889,504,819.4
	Poison Ratio, ν_{xy}	Ultimate Strength, N/m^2	Ultimate Shear Strength, N/m^2
	0.33	325,776,693.0	195,466,015.8

In conclusion, while the detailed approach using CLT was developed, it was not incorporated into the tool due to computational constraints. However, this methodology can be explored in future work, specifically when the focus shifts to composite design. The quasi-isotropic composites approach was preferred for the MM, offering a balance between accuracy and computational efficiency during the initial design stage of structural materials for novel aircraft wings.

Assumptions and Limitations in Composite Analysis

The structural sizing and mass estimation methods developed in this study offers computational efficiency and adaptability, making it a valuable tool for the conceptual or early preliminary design stages of aircraft wings. The analysis predominantly assumed quasi-isotropic composite failure when the stresses exceed allowable ultimate stress under ultimate loads for simplicity. It provides a robust platform for future research to incorporate detailed failure criteria of composite matrices and fibres.

The decision to employ the ESDU method developed for isotropic materials in shear buckling analysis is a practical choice aligned with the use of quasi-isotropic composites in the study. This approach facilitates a clearer understanding of the structural behaviour under shear stresses, acknowledging that more intricate composite analyses were not within the current study's scope. Although the presented quasi-isotropic approach is valid for initial wing mass estimation studies of SBW following Ref.(Bradley, Allen and Droney, 2015), future studies can improve the strength of the developed method by covering a more realistic analysis of composite behaviour, ensuring an improved understanding of advanced structural design considerations.

5.7 Convergent or Direct Solvers

The following chapter will explain the details of the wing box and wing group ME methods. It can be seen from Section 4.4.2.2 (Wing Mass) that an initial wing ME is required to be used in the load analysis. In other words, the wing box or wing group

masses are dependent variables of the initial wing group ME. In this study, the effect of the initial ME on the accuracy of the final result obtained from the new ME model was investigated.

As the final wing or wing box ME results are dependent variables of the initial estimation, a convergent solver approach is introduced. If the initial ME value to initialise the MM is far from the actual mass, the estimated wing or wing box masses from a direct solver might have a high error. Hence a convergent solver is generated.

The convergent solver uses the initial ME in the first iteration, and in the following iterations, it uses the ME values from the previous steps. The solver stops when the difference with the calculated wing group mass at the final iteration step and that in the previous step is below the convergent criteria introduced by the user as a percentage. The effect of direct or convergence solver will be discussed in 6.5 Validation & Comparison.

6 MASS ESTIMATION and VALIDATION

“The types of analysis used in airframe work are very different in many respects from those used in other types of engineering. The reason lies in the fact that the challenge of structural weight savings must be paramount if an efficient performance aircraft, i.e., one which carries a high percentage of payload, is to be obtained” (Niu, 1999)

In the previous chapters of the thesis, structural sizing of the load-carrying wing structures of the novel aircraft configurations is achieved. In this chapter, the mass estimation (ME) equations that rely on those sizing methods for the load-carrying box of the wing, strut, offset and jury will be presented.

In addition to the load-carrying structures, other sources of wing masses are covered in this chapter using the existing empirical methods in the literature. Several empirical methods in the literature combined with the ME method developed in this thesis. The empirical and analytical methods combinations are validated with real aircraft data. The best empirical methods fit for the new analytical methods with the lowest error values are highlighted.

As the combinations of different methods are studied, a clear group weight statement was required to avoid errors due to misinterpreting the mass data or the results of the ME methods. A detailed aircraft weight group statement (Table 1) is developed to tackle this issue and discussed in the following section.

6.1 Total Wing Group Mass

The definition of the wing group mass can be explained simply with the help of Table 1. The explanation of wing group mass in Table 1 can also help in understanding the use of the table. Wing group mass in Table 1 is located at the column of Level-4, and it can be calculated by summing the mass of the sub-components presented in Level-5 in Table 1. Level-5 sub-components of the wing group mass are the mass of the wing box, wing other mass, strut, jury-1, jury-2, and offset. Wing group mass can be formulated as given in Eq. (6-1).

$$M_{Wing\ Group} = M_{Wing\ Box} + M_{Wing\ Other} + M_{Strut} + M_{Jury-1} + M_{Jury-2} + M_{Offset} \quad (6-1)$$

Using Table 1, the sub-components of $M_{Wing\ Box}$, $M_{Wing\ Other}$, M_{Strut} , M_{Jury-1} , M_{Jury-2} , M_{Offset} can be observed. In the following sections, the mass formulas for these sub-components will be presented.

6.2 Wing Box Mass

This section will explain the formulas for the new wing box ME method. The new method combines the existing empirical methods and the analytical wing box ME method developed in this thesis. Using Table 1, $M_{Wing\ Box}$ can be formulated as given in Eq.(6-2).

$$M_{Wing\ Box} = M_{WB\ Structure} + M_{WB\ Other} \quad (6-2)$$

The details of $M_{WB\ Structure}$ and $M_{WB\ Other}$ can be seen in Table 1, and their computation will be explained in the following sections.

6.2.1 Wing Box Structure Mass

In Eq.(6-2), $M_{WB\ Structure}$ is the sum of $M_{Shear\ Total}$, $M_{Bending\ Total}$, M_{Rib} and $M_{Penalties}$ as given in Eq.(6-3).

$$M_{WB\ Structure} = M_{Shear\ Total} + M_{Bending\ Total} + M_{Rib} + M_{Penalties} \quad (6-3)$$

In this study, the structural MEs of $M_{Shear\ Total}$ and $M_{Bending\ Total}$ purely rely on analytical methods, which will be explained in the following sections. The sub-components of $M_{Penalties}$ are already covered inherently in $M_{Shear\ Total}$ and $M_{Bending\ Total}$ in the model as the concentrated and distributed inertia models are considered during the sizing process. The mass of the ribs (M_{Rib}) are covered using existing empirical methods.

6.2.1.1 Shear-Resistant Material Mass

Shear-resistant materials in a wing box section are skin panels and spar webs. The sizing process of these elements is explained in Section 5.2.1 Shear Resistant Material. In this section, the mass caused by these elements is explained. The total mass of the upper skin panels, $M_{Shear,u}$, can be computed with Eq. (6-4).

$$M_{Shear,u} = \sum_{j=1}^{N_{section}} \left(t_{Panel\ section,u,j} b_{Bay,j} \rho_{Shear} \sum_{i=1}^{\frac{N_{Boom,j}}{2}-1} w_{Panel,j,i} \right) \quad (6-4)$$

Similarly, the total weight of the lower skin panels, $M_{Shear,l,j}$, can be computed using Eq.(6-5).

$$M_{Shear,l} = \sum_{j=1}^{N_{section}} \left(t_{Panel\ section,l,j} b_{Bay,j} \rho_{Shear} \sum_{i=\frac{N_{Boom,j}}{2}}^{N_{Boom,j}-1} w_{Panel,j,i} \right) \quad (6-5)$$

The formulations to compute the total masses of the front, $M_{Web,front}$, and the rear spar webs, $M_{Web,rear}$ are given in Eqs. (6-6) and (6-7).

$$M_{Web,front} = \sum_{j=1}^{N_{section}} (t_{Web\ front,j} * b_{Bay,j} * w_{Panel,j,i}) : i = \frac{N_{Boom,j}}{2} \quad (6-6)$$

$$M_{Web,rear} = \sum_{j=1}^{N_{section}} (t_{Web\ rear,j} * b_{Bay,j} * w_{Panel,j,i}) : i = N_{Boom,j} \quad (6-7)$$

Therefore, the mass of the total shear-resistant material of the entire wing box can be computed using Eq.(6-8).

$$M_{Shear\ Total} = 2 * (M_{Shear,u} + M_{Shear,l} + M_{Web,front} + M_{Web,rear}) \quad (6-8)$$

The shear-resistant material ME method given with Eqs. (6-4)-(6-8) will be labelled as NEW-2 in the following sections. An alternative NEW-1 ME method will also be compared with NEW-2. The ME equations of NEW-1 can be derived similarly to that of NEW-2. NEW-1 differ from NEW-2 with its shear-resistant sizing approach, as explained in 5.2.1 (Shear Resistant Material).

6.2.1.2 Bending-Resistant Material Mass

The method to size the cross-sectional areas of mass booms (B_i) are explained in Section 5.2.2 (Bending Resistant Material). When the value of B_i is known the total bending resistant material mass can be calculated with Eq.(6-9).

$$M_{Bending\ Total} = M_{Stringers} + M_{Caps} \quad (6-9)$$

The formulations to compute $M_{Stringers}$ and M_{Caps} are given in Eqs.(6-10) and (6-11), respectively.

$$M_{Stringers} = 2\rho_{Bend} \sum_{j=1}^{N_{Section}} ((N_{Boom,j} - 4) * B_j * b_{Bay,j}) \quad (6-10)$$

$$M_{Caps} = 2\rho_{Bend} \left(\sum_{j=1}^{N_{Section}} 4 * B_{Cap,j} \right) * b_{Bay,j} \quad (6-11)$$

6.2.1.3 Rib Mass

It is stated by Torenbeek (1992) that the analytical ME of the ribs at the preliminary design stage is unfeasible as it requires detailed wing structural design. Hence two semi-empirical approaches from (Torenbeek, 1992; Howe, 1996) are used for the ME of the ribs, which are given in Eqs. (6-12) and (6-13), respectively.

$$M_{Rib,Torenbeek} = k_r \rho_{Rib} S_W \left(t_{Ref} + \frac{t_R + t_T}{2} \right) \quad (6-12)$$

In which S_W is the wing area, t_r and t_t are the thicknesses of the wing at the root and the tip, $k_r = 0.5 * 10^{-3}$ and $t_{ref} = 1.0$ (m) can be used as suggested by the author. The equation covers the pressure bulkheads for wing-to-fuselage connection and fuel tank boundaries.

$$M_{Rib,Howe} = \frac{3.1kS_W e(c_0 \tau_{W,0})^{0.5}}{1 + \lambda_{W,0}} \{ (1 + \lambda_{W,0} + \lambda_{W,0}^2) + 1.1c_0 \tau_{W,0} (1 + \lambda_{W,0} + \lambda_{W,0}^2 + \lambda_{W,0}^3) \} \quad (6-13)$$

Where c_0 chord length at *Section* – 0, τ_W is the thickness-to-chord ratio of the wing, λ_W is taper ratio, which is the ratio of the wing tip to the wing root chord, τ_W is the thickness-to-chord ratio at *Section* – 0, k is a factor to include the design and distribution of the ribs regarding the concentrated loads, and e is the fraction of the centre-line chord covered by the wing box. The terms k and e are suggested to be 1.3 and 0.5 by the author for simplification. The term e can also be computed using the geometric model formulated in Eq.(6-14).

$$e = \varepsilon_{rs,0} - \varepsilon_{fs,0} \quad (6-14)$$

6.2.2 Wing Box Other Masses

As can be seen from Table 1, the other sub-component of the wing box mass ($M_{Wing\ Box}$) is the wing box other mass ($M_{WB\ Other}$). It is computed using existing methods in the

literature to cover the weight additions due to aeroelastic penalties, the mass of the mounting components, and penalties due to sheet taper, joint in the panels and cut-outs.

In addition to the shear- and bending-resistant materials, non-optimum WB masses provide the complete practical WB ME by covering the corrections caused by the following reasons (Torenbeek, 1992),

- Lack of a successful optimal wing box design at all wing bays,
- Main or local joints,
- Precautions for shear lags caused by the cut-outs,
- Mountings of the concentrated loads,
- Bulkheads for wing-to-fuselage mounting,
- Weight penalties due to stiffness requirements,
- And errors due to wing box idealisations, etc.

The additional masses caused by the above-listed reasons can only be calculated by empirical methods in the preliminary design stage (Torenbeek, 1992).

The total wing box other masses can be formulated in Eq. (6-15). The formulas to compute $\Delta M_{NO,1}$, $\Delta M_{NO,2}$, and ΔW_{ST} are explained in the following sections.

$$M_{WB \text{ Other}} = \Delta M_{NO,1} + \Delta M_{NO,2} + \Delta W_{ST} \quad (6-15)$$

6.2.2.1 Sheet Taper, Joint in the Panels and Cut-outs

As previously mentioned in the structural sizing section, the theoretically minimal cross-sections for the skin panels and stringer cannot be feasible due to the increased manufacturing complexities and cost. There should be optimum cross-section changes in structural elements, and a joint is required each time a change happens. The method in (Torenbeek, 1992) can be used to estimate the additional masses caused by these joints using Eq. (6-16).

$$\Delta M_{NO,1} = \rho S_W \left(1 + 2 \frac{\bar{t}}{c} \right) \delta_{NO} / g \quad (6-16)$$

in which $\bar{t}/c$ implies the average thickness-to-chord ratio. The extra thickness for the non-optimum design is denoted by δ_{NO} which can be 10^{-3} (m) for standard built-up structures. This value can be halved for the integrally machined panels. Higher value of δ_{NO} can be used for larger cut-outs or inspection panels than the standard ones.

6.2.2.2 Mounting and Connections

Torenbeek (1992) suggests an increase of 0.1% of MTOW for wing-fuselage attachment penalty.

An increase of 0.4% MLW can be added for the weight penalty caused by the main landing gear mounting, including cut-outs, doors, ribs, doublers, and fittings. In the wing-mounted engine configuration, the mass increase caused by the pylon-attached ribs can be calculated by Eq. (6-17).

$$\Delta M_{NO,2} = 0.025(1 + 0.2N_{Eng})M_{PP}/g \quad (6-17)$$

where M_{PP} is the total concentrated mass (for wing covering pylons and nacelles) and N_{Eng} denotes the total number of concentrated loads mounted to the wing.

6.2.2.3 Aeroelastic Effects

Aeroelastic phenomena are flutter, divergence, or aileron reversal. The thickness of the wing structural elements can be increased to have a robust design against aeroelastic issues. Aeroelastic analysis requires high-fidelity models and usually causes computational loads. Unless these models are integrated into the structural sizing process, a simple empirical equation in Eq.(6-18) presented by Torenbeek (1992) following the references of (Broadbent, 1956; Sanders, 1967) can be used to account for the weight penalty to achieve a stiff enough structure.

$$\Delta W_{ST} = f(\lambda) \frac{\rho g}{G} q_D \frac{(b \cos \theta_{LE})^3 (1 - \sin \theta_{50})}{(t_{ref}/c_{ref})^2 (1 - M_D \cos^2 \theta_{50})^{1/2}} / g \quad (6-18)$$

where $\rho g/G = 10^{-6}$ (per meter) for aluminium alloys, M_D and q_D are Mach number and dynamic pressure at V_D , respectively. θ_{LE} and θ_{50} denote sweep angles at the LE and the half chord, respectively, t_{ref}/c_{ref} refers to the wing section at 70% of the half wingspan. The value of $f(\lambda)$ is 0.5 for a wing with a taper ratio between 0.3 and 0.8. For composite materials, another value can be used for the term $\rho g/G = 10^{-6}$; however, if a conservative result is aimed, this value can also be used for the composite materials.

The effect of this simple empirical approach is evaluated by comparing the data¹ obtained from Ref. (Bradley, Allen and Droney, 2015). Bradley, Allen and Droney (2015)

¹ The wing mass data in Table 24, are taken from Column F of Table 2.26 in Ref. (Bradley, Allen and Droney, 2015). The data in column F take into account the flutter constraint of $1.15 V_D$.

presented the mass values of HAR SBW Aircraft using a higher fidelity method with and without aeroelastic constraints. The effect of aeroelastic mass penalty can be extracted from this study as shown in Table 20. Hence by comparison, the effect of empirical aeroelastic mass penalty estimation on wing group mass is found -3.36% by comparing the aeroelastic mass penalty value obtained from Ref. (Bradley, Allen and Droney, 2015). This comparison shows the verification of the used method for wing group mass estimation; however, the author suggests considering aeroelastic effects with higher fidelity methods especially if UHARs are aimed to design with the presented method as will be discussed in Section 10.5.

Table 20 Comparison of aeroelastic mass penalty computed with empirical method to the reference study.

Empirical Method	Reference Study	Effect on Wing Group
Aeroelastic Penalty, kg	Aeroelastic Penalty, kg	Mass, %
474.26	249.02	-3.36

6.3 Mass of the Truss Components

The sizing procedure of the load-carrying box of the truss components was explained in Section 5.5 (Stress Analyses and Sizing of the Truss Components). The sizing procedure covers the constraints of Euler column buckling, strength at yield or ultimate stresses, bending resistance and local buckling. The procedure can be applied to all the truss components, including the strut, jury and offset.

In this study, the jury and offset are designed in one structural section as a uniform beam. Hence, their load-carrying boxes will have the same geometric dimensions and thickness values along their span. However, strut section counts depend on the jury counts as the strut geometric values can be changed between the attachments of the juries. The effects of these sections are also explained in Section - 5.5. The sizing procedure should be applied for each section of the strut separately.

After the skin thickness values of the box of the truss components are calculated using Eqs. (5-32) and (5-26), their load-carrying box mass can be computed simply using Eq. (6-19).

$$M_{Comp\ Box,j} = \sum_{j=1}^{N_{Sec}} 2t_{Comp,Skin,Selected} * (w_{Comp,box,j} + t_{Comp,box,j}) * l_{Comp,j} * \rho_{Comp\ Material} \quad (6-19)$$

In Eq. (6-19), $t_{Comp,Skin,Selected}$ is the cross-sectional area of the rectangular load-carrying box of the component, and it can be computed using Eq.(5-101). $l_{Comp,j}$ is the length of the component and $\rho_{Comp\ Material}$ is the density of the structural material of the component.

With a similar approach, the masses of the fixed LE and TE of the truss components can be computed. After finding the minimum thicknesses of the fixed LE and TE to resist the bird strikes using Eq. (5-102), their mass can be computed simply using the formulations given in Eq.(6-20).

$$M_{Comp\ LE} = \sum_{j=1}^{N_{Sec}} t_{Comp,LE,Skin,BS,j} * l_{Comp,j} * l_{Circum.\ Length} * \rho_{Comp\ Material} \quad (6-20)$$

$$M_{Comp\ TE} = \sum_{j=1}^{N_{Sec}} t_{Comp,TE,Skin,BS,j} * l_{Comp,j} * l_{Circum.\ Length} * \rho_{Comp\ Material}$$

Finally, the total mass of the truss components can be computed using Eq. (6-21).

$$M_{Comp\ Total} = 2 * (M_{Comp\ Box} + M_{Comp\ LE} + M_{Comp\ TE}) \quad (6-21)$$

It should be noted that the corrections and mounting mass shown in Table 1 were assumed to be covered with the conservative approach of the ME models developed for the jury, strut and offset. The boundary conditions can be changed from the fixed to the pinned ends to add an extra mass penalty for the strut mounting mass. Moreover, compared to the wing box, the mounting and connection, sheer taper, and cut-outs penalties are expected to be minimal on the truss components due to their structural tasks and design as a uniform beam.

6.4 Wing Other Masses

Wing other masses consist of non-load-carrying components. It can also be seen from Table 1 that wing other masses are the masses of the leading edge, trailing edge, secondary structure, landing gear doors mechanism, aileron, elevon, slats, LE flap, TE

flap, spoilers, wing speed breaks, winglets, wing folding, and pivot. The masses of those components are computed using four different methods from the references (York and Labell, 1980; Torenbeek, 1992; Chiozzotto, 2017). And each of those ME methods is combined with the mass of other wing components. These combinations of the empirical methods with the analytical method developed in this study will be evaluated in Section - 6.5.4, and the best-performing ME combination will be selected.

6.5 Validation & Comparison Studies

In the previous sections, the new ME formulations were introduced for the wing box and wing group masses of conventional and novel aircraft, including HARW, SBW, and TBW, the wings with distributed propulsion, and the wings designed for hydrogen or electric propulsions. The ME methods are validated with the available data in the literature using the dataset of 13 cantilever wings with metal wing boxes and one TBW with a composite wing box, and the results are depicted in Figure 69 and Figure 70. The aircraft data for the validation study is collected from (York and Labell, 1980; Bradley, Allen and Droney, 2015; Chiozzotto, 2016; Smith, 2016). The dataset, top and front view of the aircraft used for the validation is presented in 12.2Appendix B (Aircraft Data for Baselines and Validation).

In the validation of the new methods, in addition to using real aircraft datasets, wing group and wing box ME results from existing methods are also compared with the new methods. Other existing methods will be explained in the following section.

In Figure 69, the results of four different versions of the new wing box ME method ($M_{WB,NEW-1,Direct}$; $M_{WB,NEW-2,Direct}$; $M_{WB,NEW-1,Convergent}$; $M_{WB,NEW-2,Convergent}$) are compared. $M_{WB,NEW-2,Direct}$ and $M_{WB,NEW-2,Convergent}$ are the same as the wing box ME method explained in this thesis in the previous sections. They differ from each other with their solver. While $M_{WB,NEW-2,Direct}$ uses a direct solver and does not iterate the initial wing group ME to converge to a final solution, $M_{WB,NEW-2,Convergent}$ utilises a convergent solver to converge to a final solution. Initial wing group ME was explained in Section 4.4.2.2 (Wing Mass), and the details of the convergent and direct solvers were explained in Section 5.7.

Similarly, the difference between $M_{WB,NEW-1,Direct}$ and $M_{WB,NEW-1,Convergent}$ is their solver. While $M_{WB,NEW-1,Direct}$ utilises a direct solver, $M_{WB,NEW-1,Convergent}$ uses a convergent solver. Moreover, these two methods have different wing box skin panel sizing methods than $M_{WB,NEW-2,Direct}$ and $M_{WB,NEW-2,Convergent}$.

2,Convergent. M_WB,NEW-1,Direct and M_WB,NEW-1,Convergent have a more detailed wing panel thickness sizing approach as previously explained in Section - 5.2.1 (Shear Resistant Material). Their sizing method has different thickness values for each panel regardless of their section. On the other hand, M_WB,NEW-2,Direct and M_WB,NEW-2,Convergent methods have the same thickness value for the panels located on the same surface of the same wing box section.

Figure 70 presents the validation of the final selected wing group ME method (M_W,NEW,Selected) against the actual wing group mass values of 14 aircraft with cantilever or truss-braced wings and its comparison with the other seven existing methods. Other existing methods are explained in Section - 6.5.1 Other Mass Estimation Methods in the Literature. The details of the selected final wing group ME method are explained in Section - 6.5.4 Selecting the Best Performing Wing Group Mass Estimation Method.

The computational time of the new tool is recorded as less than 10^2 ms for one loading case with an Intel i7-11800H CPU and an octa-core 2.3 GHz computer during the validation studies.

In addition to the validation in this chapter, the new method's accuracy and sensitivity will be verified with the parametric studies in the following chapter.

6.5.1 Other Mass Estimation Methods in the Literature

The existing wing box and wing group ME methods are compared with the new methods presented in this thesis. The benefits of using the existing methods are listed below.

- The aircraft mass data was collected from several sources. Therefore, using the different existing methods can help to observe any anomalies in the collected data.
- Even if a ME method has a high accuracy level, it can still generate high errors for a specific aircraft design due to its stochastic nature. Therefore, ME with different methods could help to understand if a method failed for a specific reason.
- Comparison with other methods helps illustrate the deficiencies or advantages of the new ME method.

This study uses seven existing wing group ME and two existing wing box ME methods. In addition to the benefits listed above, the existing wing group ME methods are also

used to initialise the solver for the first iteration step as discussed in Section - 0. The studied existing wing group ME methods were previously listed in Table 13. Table 21 explains the symbols representing each existing wing group ME method in Figure 70.

Table 21 The symbols representing the existing wing group mass estimation methods.

Existing Wing Group Mass Estimation Methods	Their Symbols in Figure 70
1-P. York,1980 (York and Labell, 1980)	M_Wing,P.York
2-Chizotto,2017 (Chiozzotto, 2017)	M_Wing,Chiozzotto
3-D. Howe-Simple (Howe, 2000)	M_Wing,D.Howe
4-Raymer-General (Raymer, 2018)	M_Wing,Raymer General
5-Raymer-Transport (Raymer, 2018)	M_Wing,Raymer Transport
6-Jenkinson (Jenkinson, Simpkin and Rhodes, 1999)	M_Wing,Jenkinson
7-Cranfield Emp.	M_Wing,Cranfield Emp
8-Actual Mass	M_Wing,Actual

Table 22 The symbols represent the relevant existing wing box mass estimation methods.

Existing Wing Group Mass Estimation Methods	Their Symbols in Figure 70
1-P. York,1980 (York and Labell, 1980)	M_WBW, P. York
2-Chizotto,2017 (Chiozzotto, 2017)	M_WBW, Chiozzotto

Most existing methods are directly focused on the wing group ME. The methods for the ME of the wing box mass are relatively low in number compared to wing group ME methods. This study compares two existing wings box ME methods with the new wing box mas estimation method. The existing ME methods are from (York and Labell, 1980; Chiozzotto, 2017), and Table 22 presents the symbols to represent them in Figure 69.

6.5.2 Wing Box Mass Estimation Method Validation

The developed wing box ME methods are validated with publicly available actual data of thirteen cantilever metal wing boxes and one composite TBW box from Refs. (York and Labell, 1980; Bradley, Allen and Droney, 2015; Chiozzotto, 2016; Smith, 2016). The validation results are presented in Figure 69 with a bar chart. The aircraft's take-off gross weight (TOGW) ranges from 18.210 kg to 322.000 kg. The aircraft with aluminium alloy wing boxes are RB-66B,C-9A, C-123B, C-133B, C-135B, C-141A, DC-8, 720, 727, 737, 747, G-159 and G-1159. The TBW with composite wing box is NASA-SUGAR aircraft.

The standard deviation (SD) of the error, standard error, and average error of each wing box ME method are calculated from Figure 69 and listed in Table 23. It can be seen from Table 23 that the direct solver provides very close ME results to that of the convergent solver. As a result, considering the computational speed, the direct solver is selected as the method's default and will be used in the following sections.

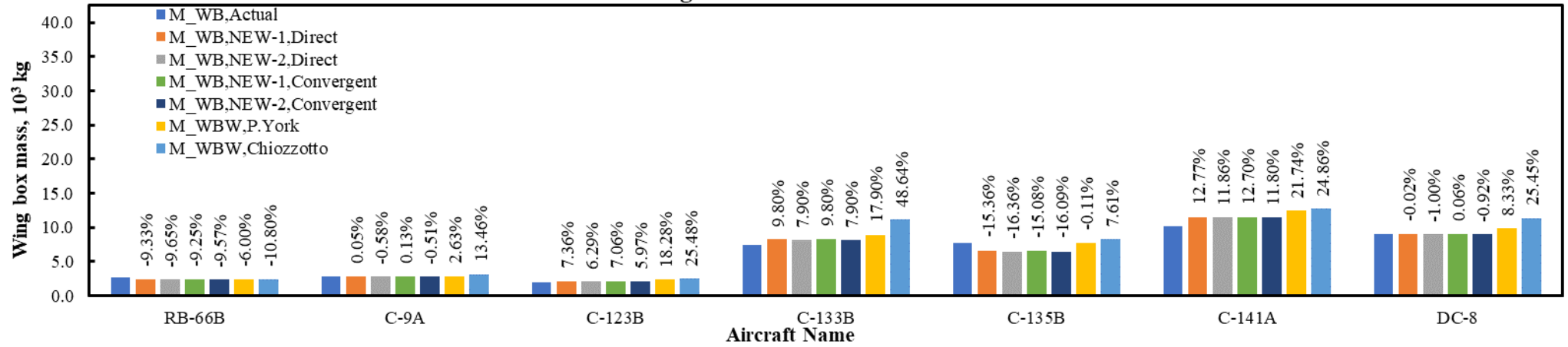
Table 23 Calculated SD of the error, standard error, and average error of each wing box mass estimation method from Figure 69.

	M_WB, NEW-1, Direct	M_WB, NEW-2, Direct	M_WB, NEW-1, Convergent	M_WB, NEW-2, Convergent	M_WBW, P. York	M_WBW, Chiozzotto
SD of the error	8.46%	8.55%	8.27%	8.36%	9.92%	20.51%
Standard error	2.29%	2.28%	2.21%	2.23%	2.75%	5.48%
Average error	1.07%	1.03%	0.24%	0.20%	9.44%	13.48%

Moreover, it can be observed that the best-performing wing box ME method with a direct solver is M_WB,NEW-2,Direct with the SD of the error, standard error, and the average error of 8.55%, 2.28%, 1.03%, respectively. Hence, M_WB, NEW-2 is selected as a default method and will be used in the following sections. M_WB, NEW-2 also achieved better estimations than the existing wing box ME methods of M_WBW,P.York and M_WBW, Chiozzotto. M_WBW,P.York achieved the SD of the error of, standard error, and the average error of 9.92%, 2.75%, and 9.44%, respectively, while M_WBW, Chiozzotto had those of 20.51%, 5.48%, and 13.48%, respectively.

The differences, which are indicated in Figure 69, between the actual wing box masses and the wing box ME of the new method might be caused by the limited publicly available aircraft design data and the sensitivity of the new tool to those parameters. The parametric study in the following chapters will explain the possible source of these differences.

Wing Box Mass Validation-1



Wing Box Mass Validation-2

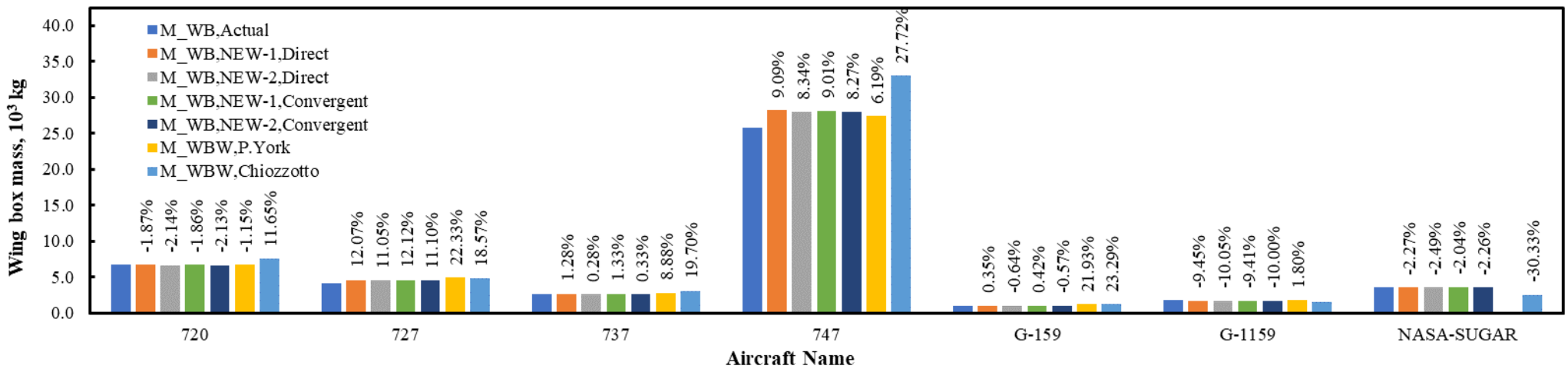


Figure 69 Validation of wing box mass estimation tool using actual data from fourteen aircraft.

6.5.3 Validation of the Mass Estimations of the Truss Components

In the design of the SUGAR TBW aircraft, higher fidelity level methods, including detailed FE analysis, computational fluid dynamics (CFD), and non-linear aeroelastic analysis, are employed compared to the new methods developed in this thesis. Hence, data² from the SBW SUGAR project (Bradley, Allen and Droney, 2015) validate the new ME methods for the truss components comprising wing, wing box, strut, and jury-1, as presented in Table 24. Besides, quasi-isotropic composite material properties are also employed from the SUGAR project (Bradley, Allen and Droney, 2015), and it was previously explained in Section - 5.6 (Structural Material).

Table 24 Detailed validation results with TBW NASA SUGAR aircraft (Bradley, Allen and Droney, 2015).

	Wing Box	Strut	Jury-1	Wing Group
Actual data	3,614.67	468.11	16.33	7,561.38
Estimation	3,533.23	395.73	14.49	7,049.37
Difference	-2.25%	-15.46%	-11.29%	-6.77%

It can be seen from Table 24 that the differences between the actual masses of the wing box, strut, Jury-1, and wing group and estimated masses of those components are - 2.25%, -15.46%, -11.29%, -6.77%, respectively. The results can be considered sufficiently accurate for conceptual and preliminary design studies.

Due to the limited publicly available SBW data from detailed design studies, the new methods are only compared with the weight data of one aircraft; however, for a better understanding of the accuracy of the new method, comparison to more aircraft is recommended when more data is available in the future. Moreover, further studies for verifying these methods with the parametric studies will be presented in the following chapter.

6.5.4 Selecting the Best Performing Wing Group Mass Estimation Method

In the previous sections, the formulas to compute the mass of the entire wing are presented. For some sub-components of the wing, more than one ME method is

² The wing mass data in Table 24, are taken from Column F of Table 2.26 in Ref. (Bradley, Allen and Droney, 2015). The data in column F take into account the flutter constraint of $1.15 V_D$.

introduced, and they are listed in Table 25 briefly. In Table 25, NEW-1 and NEW-2 are the ME methods of shear-resistant materials developed in this thesis and their difference are explained above and in Section - 5.2.1 Shear Resistant Material. The methods labelled as NEW do not have an alternative approach, and they are already explained in the thesis. Moreover, for the ME of the ribs, two empirical methods were presented in Section - 6.2.1.3 and four different methods for the wing other masses were discussed in Section - 6.4.

Table 25 The tested mass estimation methods for the relevant wing component.

Mass Group	Methods Available
Shear-resistant material mass	NEW-1 NEW-2
Bending-resistant material mass	NEW
Rib mass	1- Torenbeek (Torenbeek, 1992) 2- Denis Howe (Howe, 1996)
Wing box other mass	Torenbeek (Torenbeek, 1992)
Wing other mass	1- Torenbeek (Torenbeek, 1992) 2- York 1980 (York and Labell, 1980) 3- Chiozzotto 2017 (Chiozzotto, 2017) 4- Cranfield University
Strut mass	NEW
Offset mass	NEW
Jury mass	NEW

As a result of the combination of these different ME methods, 24 hybrid wing group ME methods are created, and they are tested with the wing group mass data of 14 aircraft, including one composite material, SBW aircraft.

The performances of the ME methods are compared with the standard deviation of the error, standard error and mean error values as depicted in Table 26. The best-performing combination is highlighted with the rectangular box and selected for the studies in the following sections.

The details of the final selected complete wing group ME method are presented in Table 27. The selected method comprises the new analytical methods for the ME of the shear- and bending-resistant materials, strut, offset and jury; the Torenbeek method (Torenbeek, 1992) for the ME of the ribs and wing box other components; and the York-1980 method (York and Labell, 1980) for the ME of wing other components.

Table 26 Combination of the wing other mass and wing rib mass calculation with the quasi-analytical wing box mass estimation method presented in this thesis.

WB Options	Error Types	M_WSec, Torenbeek	M_WSec, P.York	M_WSec, Chizotto	M_WSec, Cranfield
M_WB,NEW-1 + M_Rib_Tor	SD of the error	11.56%	6.60%	8.46%	5.92%
	Standard error	3.09%	1.76%	2.26%	1.58%
	Average error	-11.13%	-1.71%	2.71%	-16.06%
M_WB,NEW-1 + M_Rib_D.Howe	SD of the error	12.90%	7.99%	10.10%	7.43%
	Standard error	3.45%	2.14%	2.70%	1.99%
	Average error	-12.54%	-3.12%	1.29%	-17.47%
M_WB,NEW-1 + M_Rib_Cran.	SD of the error	14.06%	9.59%	12.00%	8.52%
	Standard error	3.76%	2.56%	3.21%	2.28%
	Average error	-6.13%	3.29%	7.70%	-11.06%
M_WB,NEW-2+ M_Rib_Tor	SD of the error	11.39%	6.49%	8.25%	5.83%
	Standard error	3.04%	1.74%	2.21%	1.56%
	Average error	-11.64%	-2.22%	2.19%	-16.57%
M_WB,NEW-2+ M_Rib_D.Howe	SD of the error	12.74%	7.88%	9.92%	7.34%
	Standard error	3.40%	2.11%	2.65%	1.96%
	Average error	-13.06%	-3.63%	0.78%	-17.99%
M_WB,NEW -2+ M_Rib_Cran.	SD of the error	13.90%	9.49%	11.84%	8.43%
	Standard error	3.72%	2.54%	3.16%	2.25%
	Average error	-6.65%	2.78%	7.19%	-11.58%

Table 27 The complete hybrid wing group mass estimation method and its components.

Mass Group	Selected Method
Shear-resistant material mass	NEW-2
Bending-resistant material mass	NEW
Rib mass	Torenbeek (Torenbeek, 1992)
Wing box other mass	Torenbeek (Torenbeek, 1992)
Wing other mass	York 1980 (York and Labell, 1980)
Strut mass	NEW
Offset mass	NEW
Jury mass	NEW

6.5.5 Wing Group Mass Estimation Method Validation

In the previous section, the best-performing wing group ME method is selected among 24 hybrid ME methods. In this section, the details of the validation of the selected wing group ME method (M_Wing, NEW, Selected or M_Wing, NE-2) will be presented, and its performance will be compared to seven existing methods from the literature.

The wing group ME from (York and Labell, 1980) is labelled M_Wing, P.York in Table 28. It should be noted that this semi-empirical wing group ME method is developed

utilising the aircraft dataset used for the validation in this section. The validation performance of M_Wing, P.York is expected to be high as its regression samples are used for the validation. Hence any method performing better than M_Wing, P.York in this validation study would be considered with high credibility.

Table 28 Comparison of errors of wing group mass estimation of different methods.

	M_Wing, NEW, Selected	M_Wing, P.York	M_Wing, Chiozzotto	M_Wing, D.Howe
SD of the error.	6.49%	7.52%	14.55%	15.29%
Standard error	1.74%	2.09%	3.89%	4.24%
Average error	-2.22%	3.76%	11.78%	-16.97%
	M_Wing, Raymer General	M_Wing, Raymer Transport	M_Wing, Jenkinson	M_Wing, Cranfield Emp
SD of the error.	12.08%	13.70%	19.33%	15.81%
Standard error	3.35%	3.80%	5.36%	4.39%
Average error	-25.99%	-16.34%	-43.80%	-6.10%

The mass group MEs of M_Wing,NEW,Selected (M_Wing,NEW-2) and seven existing methods for 13 metal cantilever wings and one composite TBW are depicted in Figure 70. The error of ME of each method is presented above the bars. The aircraft in Figure 70 is the same as that used for the wing box ME validation study. Therefore, the aircraft's take-off gross weight (TOGW) ranges from 18.210 kg to 322.000 kg. The aircraft with aluminium alloy wing boxes are RB-66B, C-9A, C-123B, C-133B, C-135B, C-141A, DC-8, 720, 727, 737, 747, G-159 and G-1159. The TBW with composite wing box is a NASA-SUGAR aircraft. The details of the validation data is given in 12.2Appendix B - (Aircraft Data for Baselines and Validation).

The standard deviation (SD) of errors, standard error and average error of all ME methods are shown in Table 28. It can be observed that the new method developed in this thesis presented the best performance with an SD of errors of 6.49%, standard error of 1.74, and average error of -2.22%. The second best performing method is M_Wing,P.York.

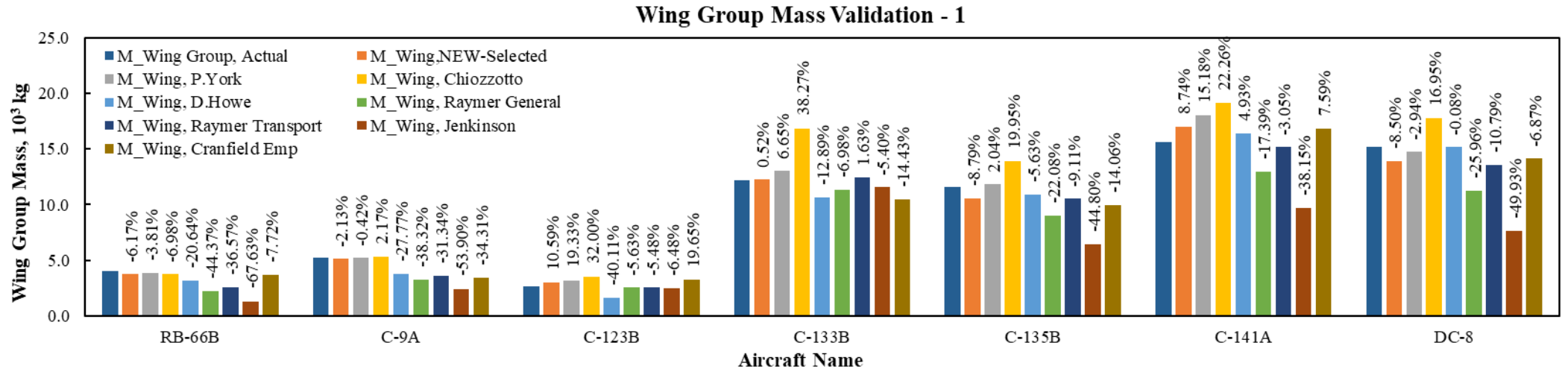
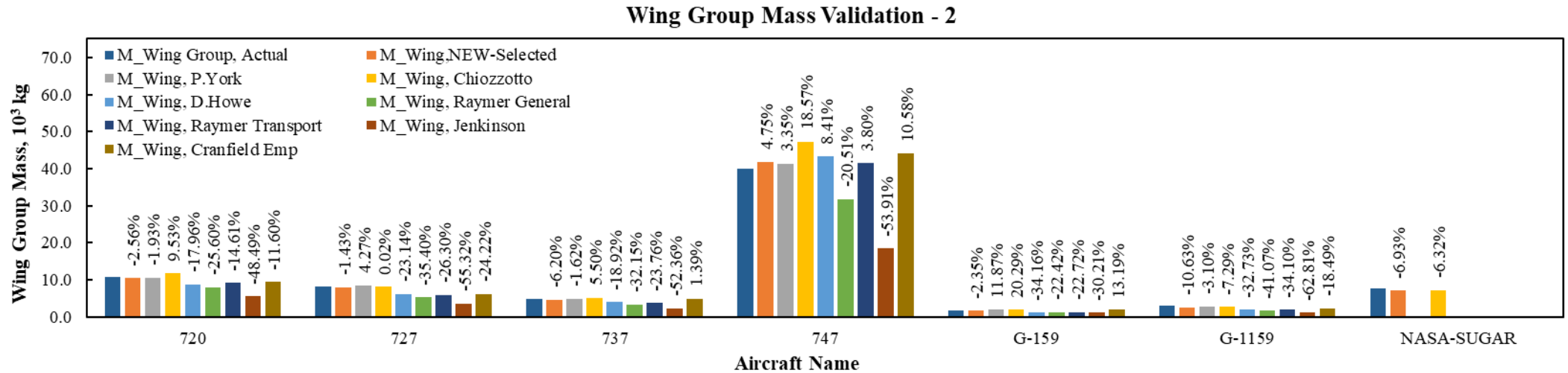


Figure 70 Validation of the wing group mass estimation method using the actual data of fourteen aircraft.

In conclusion, these validation studies confirm the accuracy level of the new wing group mass and wing box ME methods developed in this thesis. The new methods had the best performance in the validation studies compared to other methods. After validating them with 13 cantilever metal wing aircraft and 1 TBW aircraft, their credibility is increased to be employed in other research. In the following chapters, their sensitivity and the results of parametric analyses are presented and discussed. These studies help increase the credibility of the new methods and can be used to verify the methods for the novel configurations.

6.5.6 Comparison of the Methods with Carpet Plots

This section strives to expose the behaviour of traditional ME methods and the developed method, particularly in the context of increasing ARs - extending from short AR wings to UHARW.

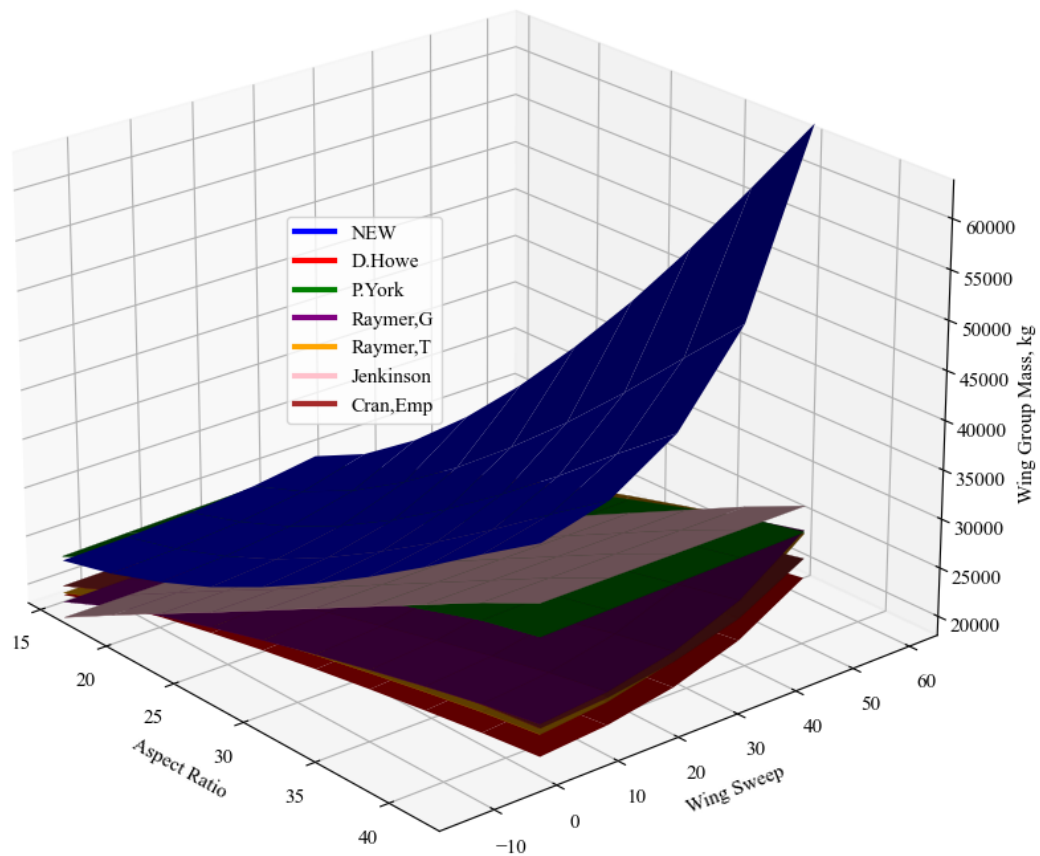
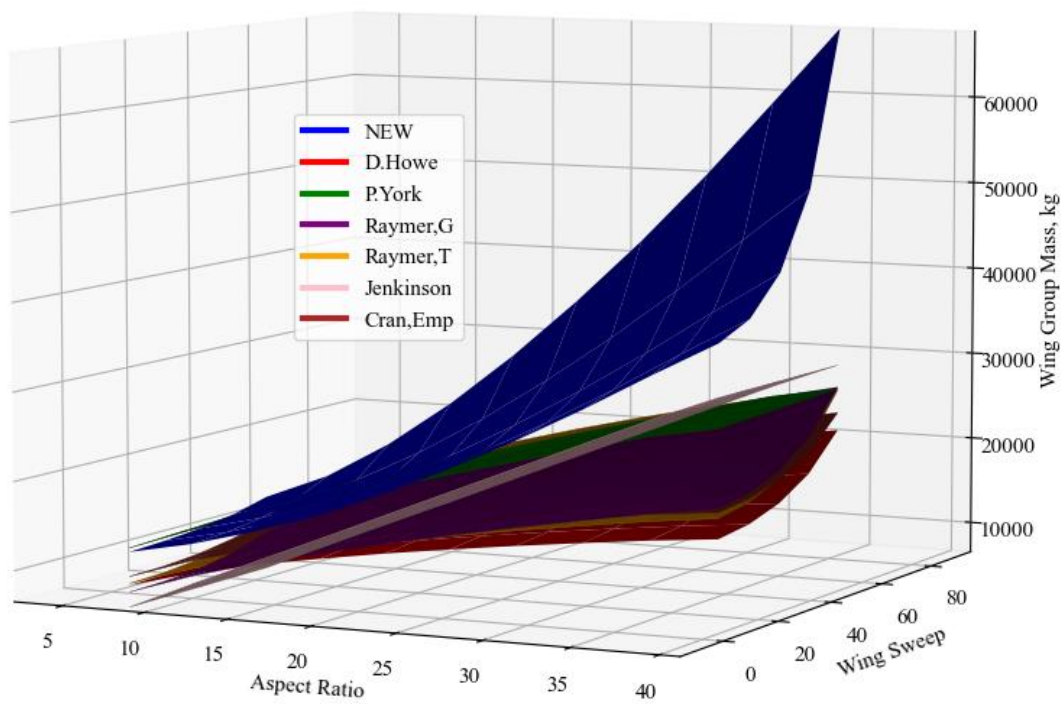
It was hypothesized that a novel ME method, based on analytical structural sizing and ME (SSaME) of load-carrying structures, would exhibit enhanced sensitivity to critical aircraft design parameters, including AR. This study stands as a testament to the validation of the hypothesis.

Thus, the section unfolds as follows:

The novel wing group ME methodology is verified by juxtaposing it with prevalent methods in the field. This comparative analysis is visually represented through 3D carpet plots (refer to Figure 71). These plots, a reflection of the hypothesis, facilitate an assessment of the developed methodology's response to variations in key parameters of wing design.

To construct these carpet plots, two key variables, namely the wing AR and sweep angles were manipulated, while maintaining other variables, such as the wing taper ratio and wing surface area, constant. The variation in AR was achieved by adjusting the wingspan and wing root chord values.

The Boeing 720 aircraft, detailed specifications of which are provided in Table 56, served as the baseline model for this study. The table indicates that the baseline model's AR and wing leading edge (LE) sweep angle are 7.03 and 35.77 degrees, respectively. As illustrated in Figure 70, the novel ME method calculated the wing group mass of the Boeing 720 with an error of merely 2.56%.



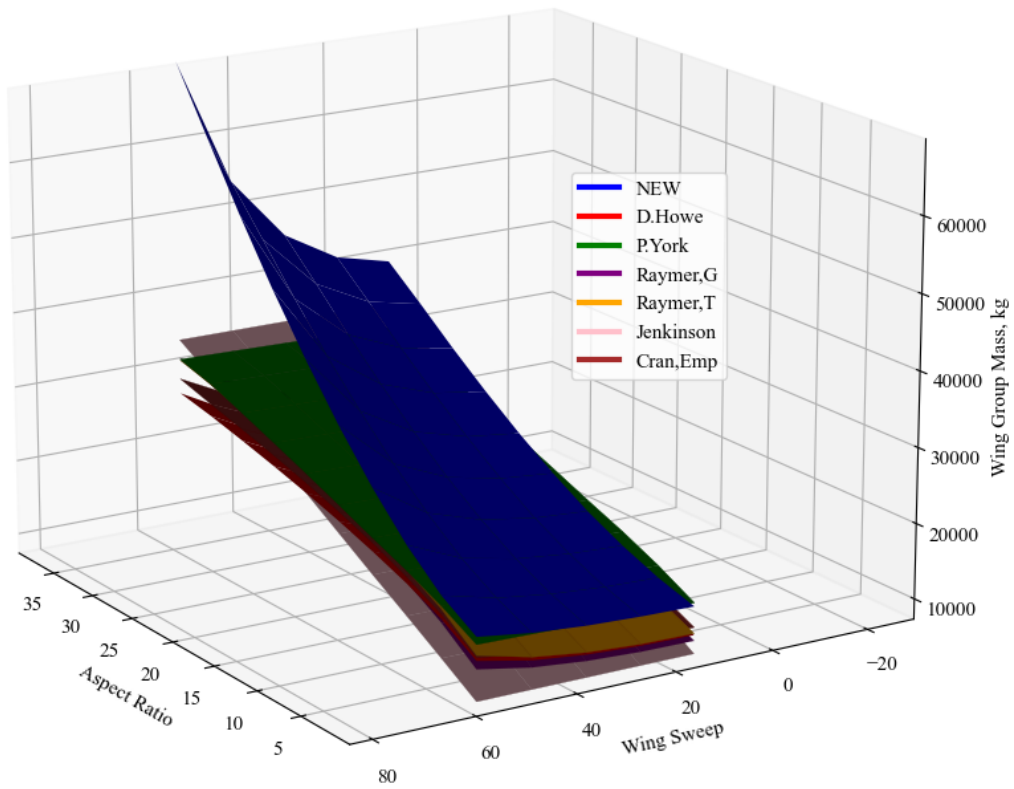
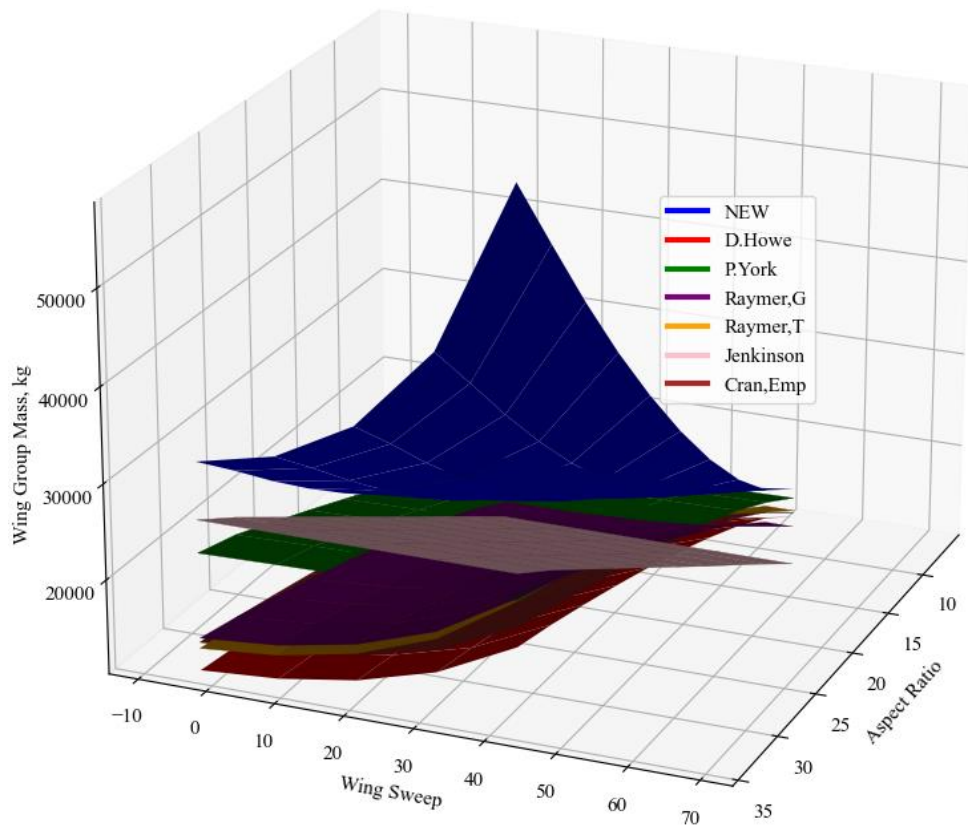


Figure 71 Four perspectives of the 3D carpet plot showcasing variations in wing sweep, aspect ratio, and computed wing group mass.

The carpet plots in Figure 71 illustrate the heightened sensitivity of the new method to alterations in the wing LE sweep angle and AR. This enhanced sensitivity provides a critical advantage in the prediction of UHARWs. In contrast, existing ME methods, largely grounded in semi-empirical principles, heavily depend on historical aircraft data. This dependence impairs their capability to predict the mass of wings with ultra-high ARs accurately. This study has not evaluated Chiozzotto's (2017) method due to its failure to handle extreme design variable values.

The proposed method, conversely, utilises a higher fidelity analytical approach for SSaME of load-carrying components. This approach accurately reflects the increased bending loads associated with high AR values.

Figure 71 clearly illustrates that all methods' wing group mass estimates are similar for AR values less than 10. This unity can be attributed to the semi-empirical nature of existing methods, which rely predominantly on historical data of aircraft with moderate AR wings.

However, a divergence in ME becomes evident beyond an AR of 10. The estimations from existing methods either increase linearly or with a diminishing gradient, while the new method reflects an increasing gradient as AR values exceed 10.

In summary, the behaviour of the new model aligns with the expectations of a mass increase concurrent with rising AR. Although the detailed validation of the developed MM in the ME of the HARW is provided in the previous sections, a detailed validation in the UHARW is still suggested in future studies when real data for this type of wing configuration is available.

7 PARAMETRIC and SENSITIVITY STUDIES

This chapter delves into the results of a detailed parametric and sensitivity analysis, elucidating the implications of various aircraft design parameters on the wing box mass. The chapter is divided into two core sections, each offering a unique perspective on the interplay between aircraft design parameters and the masses of the novel wing components.

The first section investigates the influence of design parameters of the wing box, inertia, and wing planform on the wing box masses exclusively (Taflan, Smith and Loughlan, 2023). This analysis will draw upon four distinct baseline aircraft, each exemplifying moderate or HAR cantilever or TBWs. By comparing these diverse aircraft designs, it is aimed to offer an intricate understanding of how different design parameters of different configurations impact the wing box mass. Moreover, by using an importance factor in the first section, the impact of each design parameter was measured and compared, providing an elaborative understanding of their relative importance. The second section provides an in-depth parametric analysis of a TBW aircraft. It investigates how the design parameters affect the wing box, wing group, strut, truss, offset, and jury masses.

By the end of this chapter, readers should gain a comprehensive understanding of the complex relationships between aircraft design parameters and the masses of the wing components of the studied aircraft. This understanding is crucial for future aircraft design optimization and developing more efficient, lighter, and environmentally friendly aircraft.

7.1 Four Baseline Aircraft

For the comparative and parametric studies, four aircraft are chosen as baselines, including the Boeing 737, the Boeing 747 (York and Labell, 1980; Chiozzotto, 2016; Smith, 2016) and NASA's SUGAR-SBW aircraft (Bradley, Allen and Droney, 2015) as well as a cantilever version (SUGAR-CANT) of the NASA SUGAR-SBW aircraft. Apart from the absence of a strut in SUGAR-CANT, the other configurations are the same as SUGAR-SBW. It should be noted that although the SUGAR-SBW configuration has a jury, it is named SUGAR-SBW instead of TBW in this chapter. Consequently, the quartet of configurations comprises SUGAR-SBW, SUGAR-CANT, Boeing-737, and Boeing-747. Detailed design parameters of these baseline models are presented in Table 56 and Table 57.

The choice of the SUGAR-SBW and SUGAR-CANT configurations enables an investigation into the behaviour of HAR cantilever and Strut-Braced Wing (SBW) aircraft and facilitates comparison with moderate AR aircraft, exemplified by the Boeing 747 and Boeing 737.

The primary distinction between the Boeing 747 and Boeing 737 lies in their dimensions, TOGW and OEW. Notably, the Boeing 747 is the sole configuration among the selected baseline aircraft to feature four wing-mounted engines, while the remaining configurations are equipped with two wing-mounted engines.

7.2 Wing Box Parametric Study

This section presents a robust parametric study investigating the impact of fourteen distinct aircraft design parameters on the total wing box structural mass. Comparisons are made with prior studies, providing insights into the new ME method's capabilities.

The investigation assesses the influence of aircraft design parameters on the wing box weight of moderate and HARWs, using reference points such as SUGAR-SBW, SUGAR-CANT, Boeing 747, and Boeing 737. The presence of a strut and its impact on design parameters' sensitivity is also examined.

The OAT method is employed for the parametric study. It alters one input parameter at a time, enabling the evaluation and comparison of each parameter's effects. However, the study compares four aircraft with varied TOGW, ARs, and configurations to account for simultaneous parameter changes.

The following sub-section begins with a comparison of design parameters using an importance factor, followed by a study of the strut position's impact on the wing box mass of SBW configurations. Finally, design parameters are divided into three categories: wing box design parameters, parameters related to inertia, and wing planform design parameters.

7.2.1 Comparison of All Input Parameters

To measure the sensitivity of wing box weight to each input parameter across different configurations, an importance factor is applied, as proposed by Solano et al. (2021) (see Eq.(7-1)). This factor utilizes the maximum and minimum wing box masses corresponding to individual parameter changes. The range of these parameter changes

is delineated in the forthcoming graphs. The calculated importance of each parameter for all configurations is presented in Table 29 together with the lower and upper bounds.

Table 29 The importance of aircraft design parameters or the sensitivity of the wing box mass to those parameters.

	SUGAR-SBW		SUGAR-CANT		Boeing-737		Boeing-747	
Studied Design Parameter	Up. B. - Low. B.	Importance (%)	Up. B. - Low. B.	Importance (%)	Up. B. - Low. B.	Importance (%)	Up. B. - Low. B.	Importance (%)
Stringer Count	3-31	21.6	3-31	19.2	3-31	30.9	3-31	27.6
WB Shape Change	0-1	41.3	0-1	90.2	0-1	30.5	0-1	43.5
ND Distance Between the Spars	0.1-1	124.3	0.1-1	38.0	0.1-1	90.0	0.1-1	87.8
Centre Wing Box Length	1.2-8 (m)	39.9	1.2-8 (m)	0.7	0.2-10 (m)	20.2	0.5-15 (m)	3.8
TOGW	60-120 (Tonne)	40	60-120 (Tonne)	72.73	45-105 (Tonne)	22.22	300-600 (Tonne)	40.89
OEW	5-50 (Tonne)	3.05	5-50 (Tonne)	3.05	5-25 (Tonne)	1.4	50-150 (Tonne)	0
Fuel Stored in the Wing	0-31 (Tonne)	18.1	0-31 (Tonne)	28.9	0-30 (Tonne)	7.8	0-170 (Tonne)	12.8
Wing Group Mass	1-25 (Tonne)	22.7	1-25 (Tonne)	31.0	5-15 (Tonne)	5.0	6-80 (Tonne)	5.5
ND Spanwise Engine Position	0.1-0.95	18.0	0.1-0.95	37.9	0.1-0.95	9.1	0.1-0.95	5.7
Root Thickness to Chord Ratio	0.04-0.29	38.0	0.04-0.29	107.1	0.04-0.29	44.8	0.04-0.29	242.4
Tip Thickness to Chord Ratio	0.04-0.29	60.3	0.04-0.29	64.7	0.04-0.29	38.5	0.04-0.29	38.8
Aspect Ratio	2.5-40	348.8	2.5-40	325.3	2.5-35	392.6	2.5-40	274.5
ND Spanwise Position of the Strut	0.1-1	34.2						
ND Chordwise Position of the Strut	0.1-0.95	3.2						

$$\%Importance = \frac{Maximum\ WB\ Mass - Minimum\ WB\ Mass}{Average\ of\ WB\ Masses} \times 100 \quad (7-1)$$

It can be seen from Table 29 that the wing box masses of all configurations are sensitive to the AR the most among the fourteen parameters presented, and the sensitivities of WB masses of different configurations to the same input parameter are not the same. In the following sections, a deeper discussion of the effect of the design parameters will be provided, interpreting Table 29 with more detailed graphs.

7.2.2 Strut Attachment Position

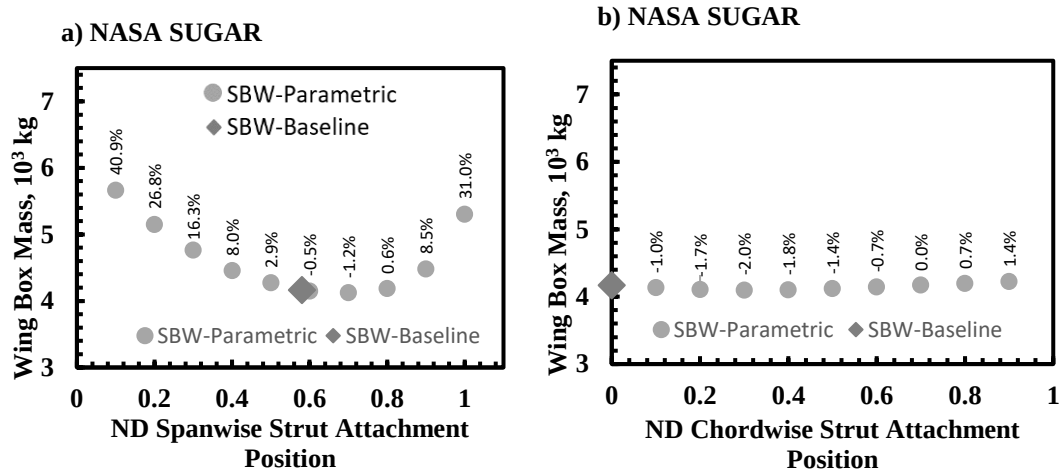


Figure 72 NASA SUGAR baseline aircraft wing box mass variation with a) spanwise ND strut attachment point change as a fraction of wing half span and b) chordwise ND strut attachment point change as a fraction of wing section chord.

The position of the strut attachment is an important design parameter for SBW aircraft, and it has been studied previously in Refs. (Bhatia *et al.*, 2009; Chiozzotto, 2016). The sensitivity of the new wing box ME method to the strut attachment point is presented in Figure 72. For the SUGAR-SBW baseline, the minimum WB mass is observed at an ND spanwise position of 0.65. The optimum position of the strut is located marginally outboard compared to previous studies, as the current work only presents the structural mass of WB. When the strut mass is added to the WB mass, the optimum point is likely to move slightly inboard, as justified in Section 7.4.1.1.

It can also be observed that WB mass is less sensitive to the ND chordwise position of the strut compared to the ND spanwise position. The optimum attachment point for the minimum WB mass tends to be towards the wing's leading edge.

7.2.3 Wing Box Design Parameters

7.2.3.1 Stringer Counts

The stringer count in Figure 73 shows half of the total stringer counts in a wing box section. The importance of the stringer count is almost equal for all configurations, as shown in Table 29, but its trend is different for moderate and HARWs, as illustrated in Figure 73. This difference can be explained by the effect of the different dominant loads for different configurations. With increasing AR (SUGAR-SBW, SUGAR-CANT), the bending loads are becoming more dominant in structural sizing, and the bending-resistant materials cause the major portion of the wing box mass. In contrast, the shear-resistant materials are a major part of the wing box mass in moderate aspect ratio Boeing 737 and 747.

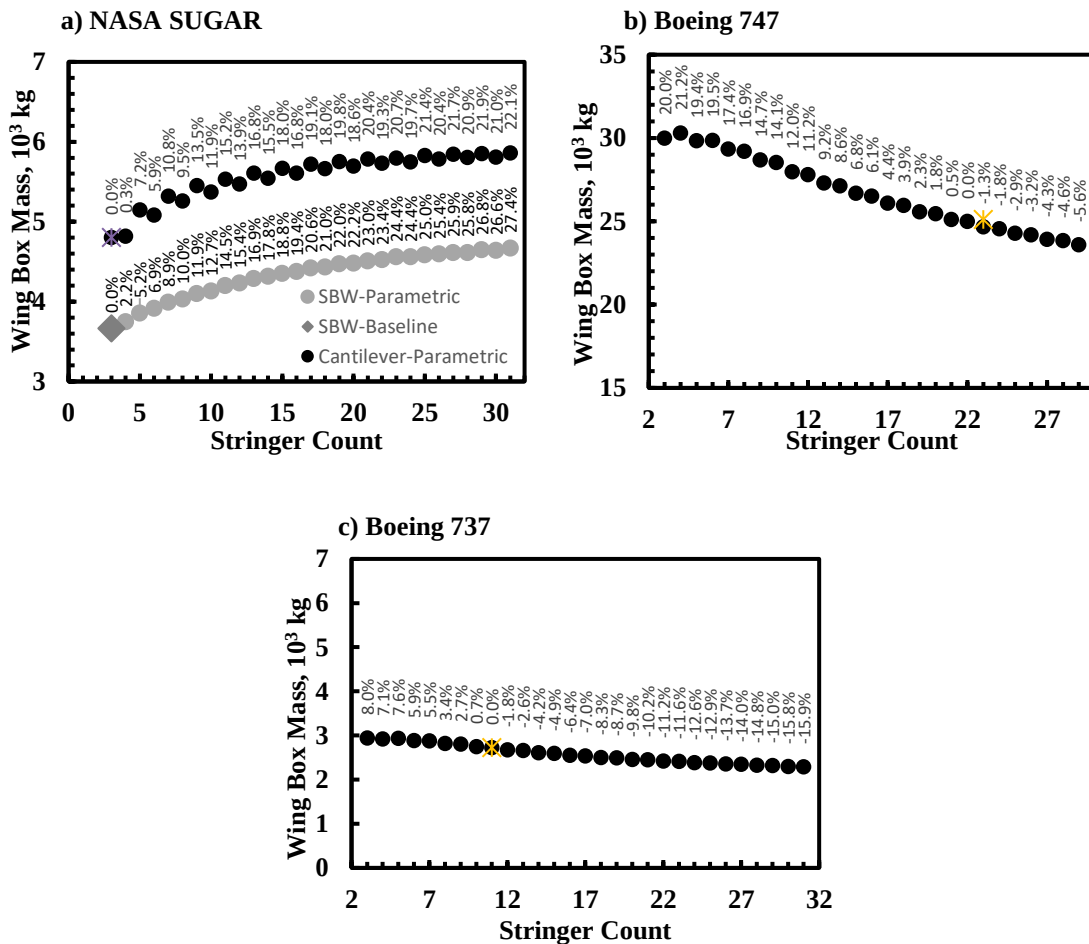


Figure 73 a) NASA SUGAR-SBW, SUGAR-CANT; b) Boeing 747; c) Boeing 737: baseline aircraft wing box mass variation with total stringer counts.

In moderate AR wings (Boeing 747 and 737), however, the effect of shear loads is more dominant, and shear-resisting materials account for a significant portion of the WB mass. Hence, if the number of stringer counts is increased in a moderate AR wing, the distance between the stringers is reduced, preventing plate buckling and reducing the amount of shear-resistant material used. The detailed wing box mass breakdown of the studied baseline aircraft is given in Appendix 12.2B.3.

7.2.3.2 ND Distance Between the Spars

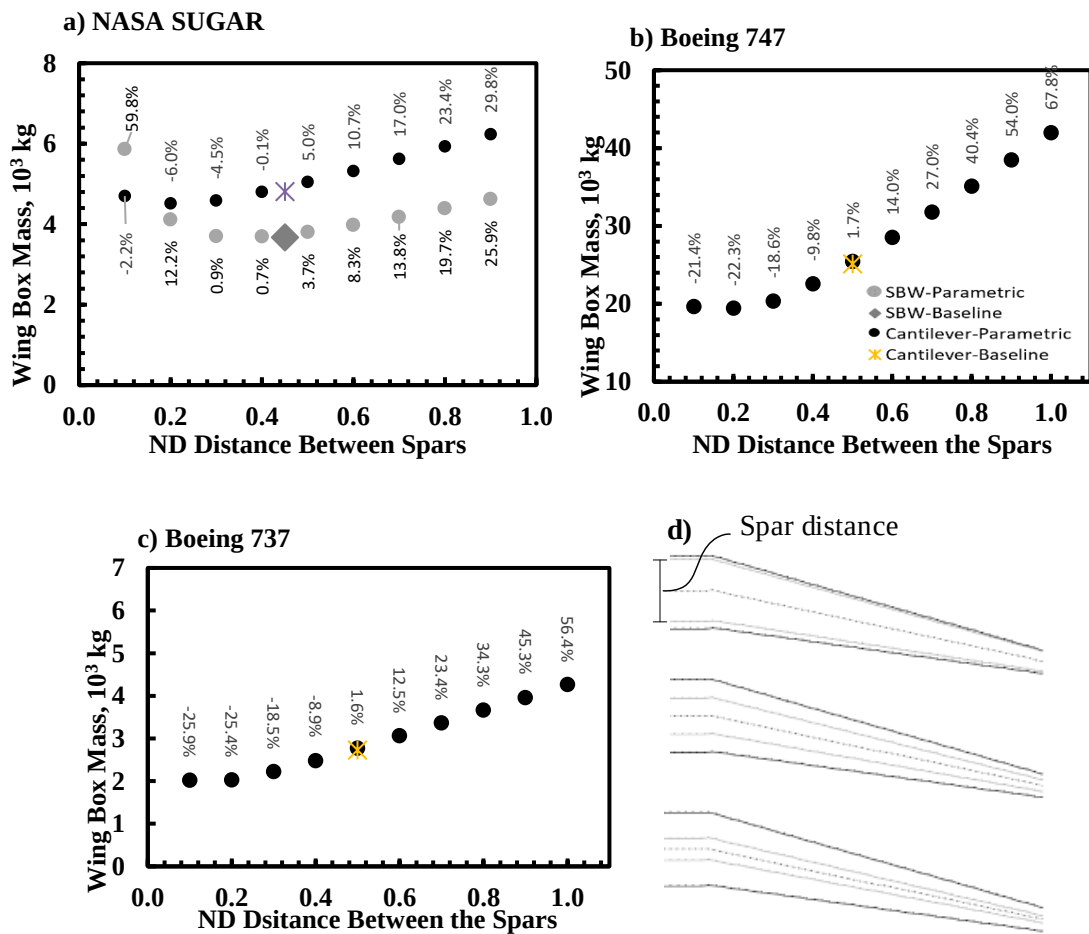


Figure 74 a) NASA SUGAR-SBW and SUGAR-CANT, b) Boeing 747, c) Boeing 737, baseline aircraft wing box mass changes as ND distance between forward and rear spars as a fraction of wing section chord changes, d) illustration of the distance change between the spars.

Apart from that, in HARWs, the bending loads are more dominant, and increasing the number of stringers does not provide the same benefit for weight reduction. Due to the assumption of the same mass boom cross-section in a wing box section in the new model, increasing the number of stringers increases the total bending material mass, as

all the mass booms are sized with the same cross-sectional area as the thickest mass boom cross-section calculated in that section. This assumption causes oversized mass-boom components in the model. In summary, it can be said that a higher number of stringers provides more benefits for wings with a moderate AR compared to HAR ones.

The ND distance between the spars is a fraction of the wing section chord. The WB masses of the SUGAR-CANT, Boeing 737, and Boeing 747 increase with increasing ND distance between the spars, as shown in Figure 74. However, the SUGAR-SBW had an optimum value around the ND value of 0.4. The buckling constraint can explain the increase in WB mass as the distance between the spars increases. When the ND value reaches one, the distance between the spars increases with the fixed stringer count, which causes wider skin panels and requires thicker skins to resist the buckling.

The trend of the reactions of all configurations to ND spar distances is similar; however, their sensitivity levels are different, as can be observed in Table 29. With an importance factor of 124.3, SUGAR-SBW has the highest sensitivity to this parameter. Hence, it can be said that for the conceptual design studies of novel configurations, this parameter should be studied as a design variable. Moreover, the optimum point for the SUGAR-SBW aircraft's spar positions is more distinguishable than the other configurations.

7.2.3.3 Idealized Wing Box Shape

The idealized wing box shape variation is previously illustrated in Figure 18. When the non-dimensional wing box edge thickness parameter ($\Psi_{WB,Edge,j}$) is increased from 0 to 1, and the wing box shape of all configurations changes from a diamond to a hexagonal and finally to a rectangular shape. It can be seen from Figure 75 that the wing box masses of all four configurations are reduced when the wing box shape approaches to a rectangular from a diamond shape. Moreover, all configurations are very sensitive to the wing box shape, as presented in Table 29. With a rate of importance of 90%, the SUGAR-CANT aircraft wing is the most sensitive configuration to this parameter.

Although a rectangular wing box shape (for $\Psi_{WB,Edge,j} = 1$) seems ideal for a light wing box mass; it is usually not feasible for most design cases due to the airfoil shape. As a result, a hexagonal model, as shown in Figure 18, is more representative of the actual wing box cross-section. The WB shape can also approach a rectangular shape when the distance between the spars is reduced. It is shown in the previous section that the wing box mass tends to reduce with the decrease between the wing box spars. Thus, the combined effect of the reduced distance between the spars and the rectangular WB

shape can provide further WB mass reduction; this outcome should be studied in more detail. This design approach may be promising for future hydrogen-powered aircraft wings because fuel will probably not be stored in the wing box, and the distance between the spars can be adjusted without fuel tanks.

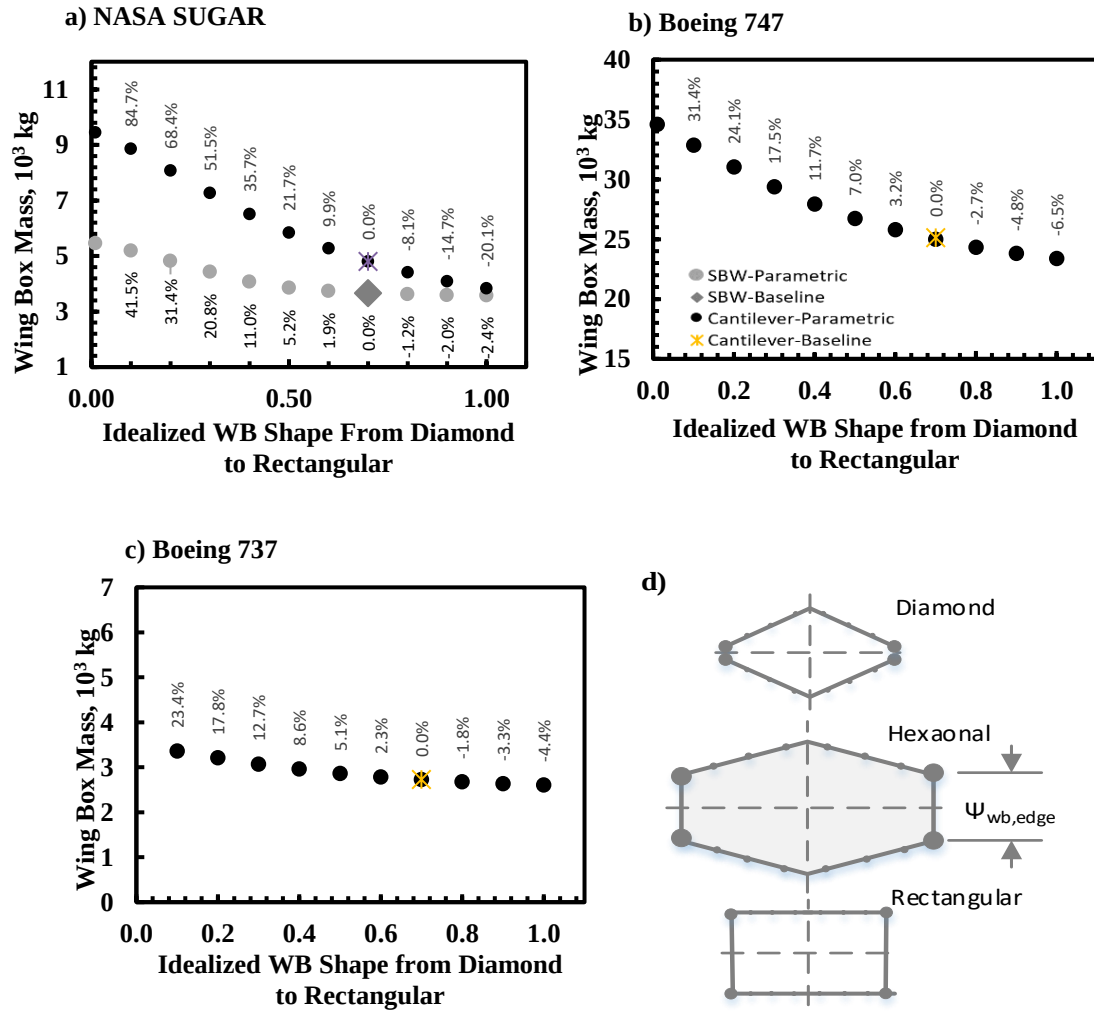


Figure 75 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737: the baseline aircraft wing box mass varies as the ND wing box edge thickness value (ψ_{edge}) changes, d) the illustration of the wing box shape changes from a diamond to a hexagonal and a rectangular with the change of $\psi_{WB,Edge,j}$.

7.2.3.4 Centre Wing Box Length

The effect of the length of the centre wing box on the WB mass is independent of the AR, as seen in Figure 76 and Table 29. While the SUGAR-SBW aircraft and Boeing 737 are the most sensitive configurations to this parameter, with an importance factor of

39.9% and 20.2%, respectively, the SUGAR-CANT aircraft and Boeing 747 have minimal sensitivity.

Although SUGAR-SBW and the Boeing 737 are sensitive to the length of the centre wing box, they have different tendencies. While the increased length of the centre wing box reduced the WB weight of the Sugar SBW by 9.96%, it increased the WB weight of the Boeing 737 by 17.8% when compared to the baseline values. The weight reduction in SUGAR-SBW is likely to be due to the torsional relief caused by the strut reaction; however, the increase in the WB weight of the Boeing 737 can be due to the increased torsion on the centre wing box of this configuration. To better understand the effect of this parameter, further studies are required.

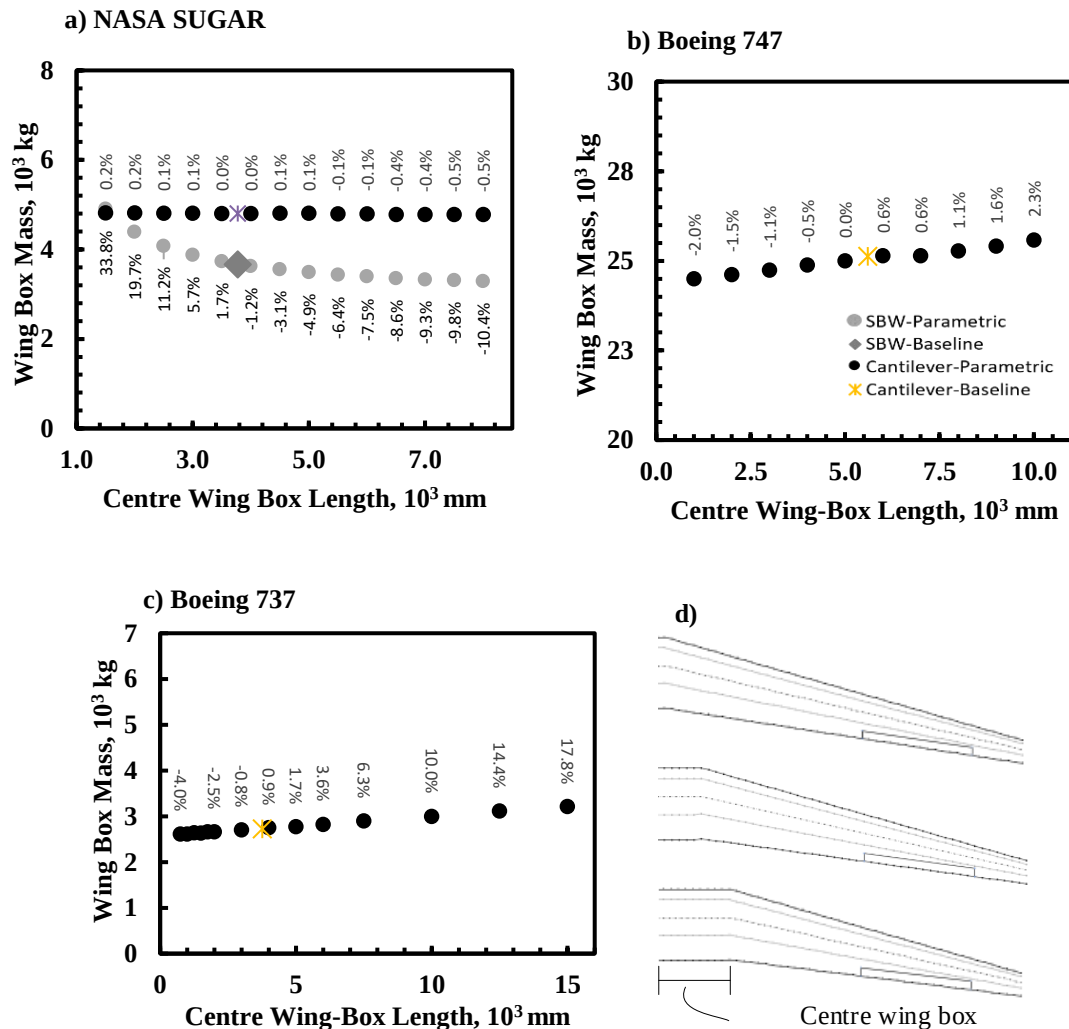


Figure 76 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737: baseline aircraft wing box mass changes with the change of the centre wing box length, d) illustration of the centre wing box length change.

7.2.4 Inertia

7.2.4.1 Take-off Gross Weight and Operating Empty Weight

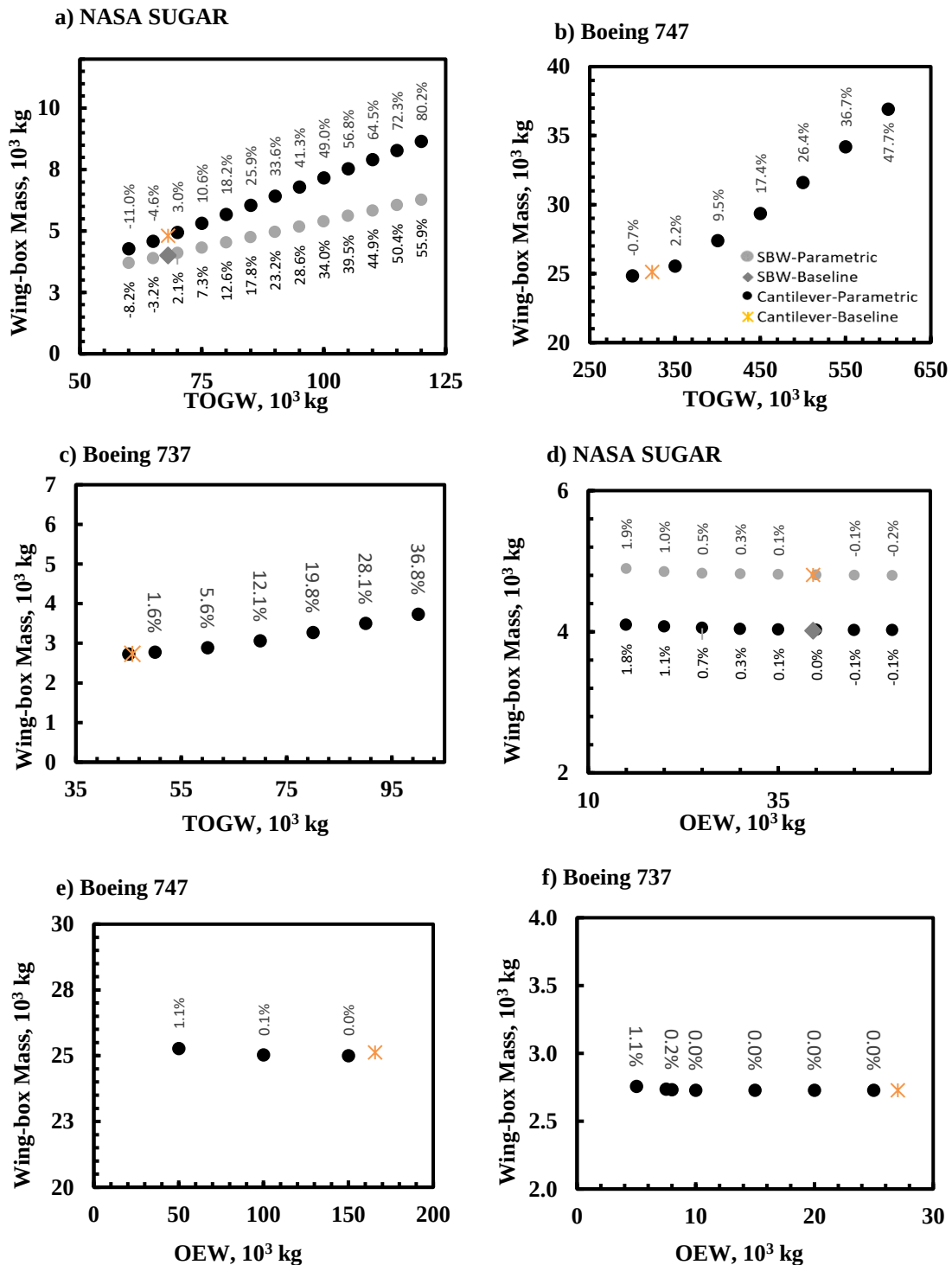


Figure 77 a) NASA SUGAR-SBW, SUGAR-CANT; b) Boeing 747, c) Boeing 737: baseline aircraft wing box mass changes with the change in the aircraft take-off gross weight (TOGW). d) NASA SUGAR-SBW, SUGAR-CANT, e) Boeing 747, f)

Boeing 737: As the aircraft take-off gross weight (OEW) changes with changes in the baseline aircraft wing box mass.

The sensitivity of the wing box masses of all configurations to the TOGW and OEW are shown in Table 29. While the WB masses of all configurations are sensitive to TOGW, the OEW has a slight effect on the WB masses. The increase in WB weight with the increases in TOGW (shown in Figure 77) is expected as it might be explained by the increased loads and lift acting on the wing. Figure 77 shows the importance of the TOGW parameter on the wing mass. Although the change in OEW affects the available capacity for the fuel and payload, the effect of OEW on the wing box mass appears to be minimal for the studied baseline aircraft.

7.2.4.2 Fuel Stored in the Wing

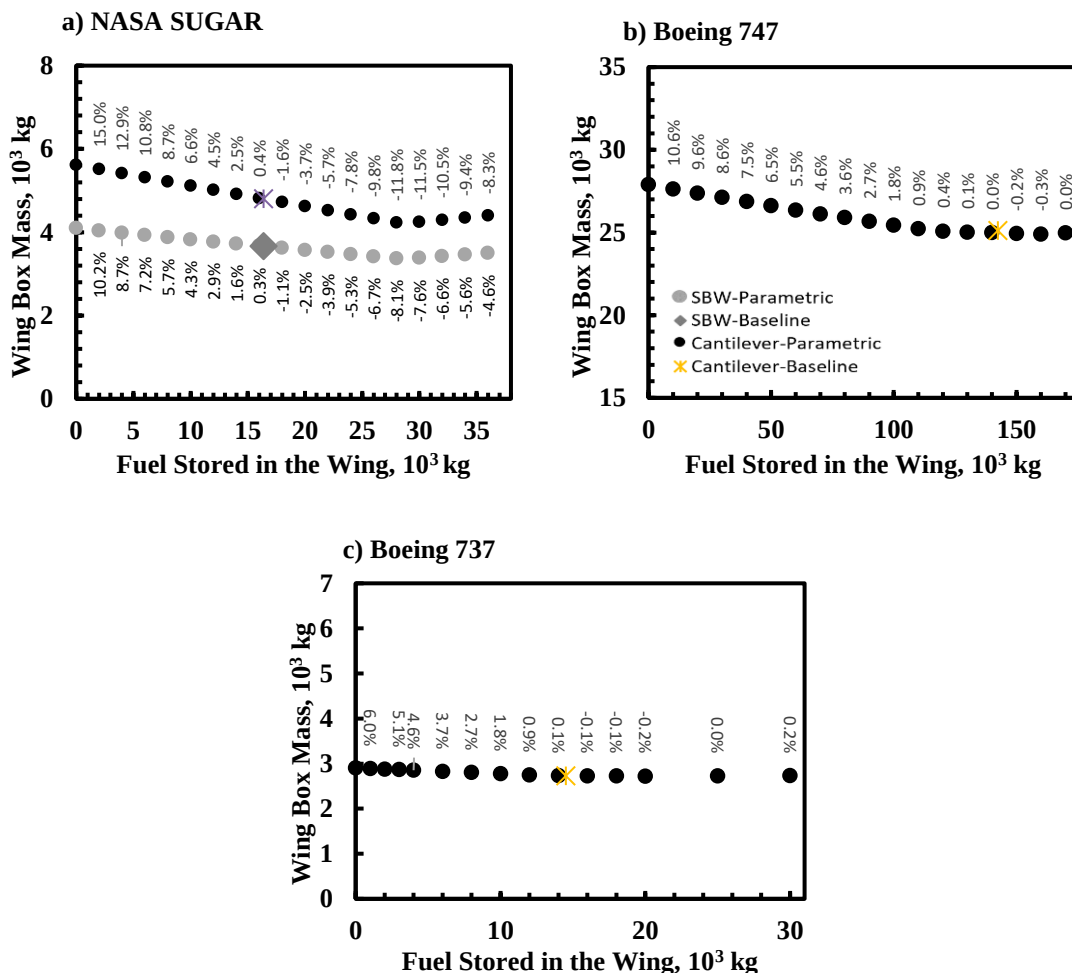


Figure 78 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737: baseline aircraft wing box mass change with the change of the fuel mass stored in the wing.

It can be seen from all the graphs in Figure 78 that the fuel stored in the wing provides inertia relief for the WB and results in a mass reduction of the WB structure. HAR SUGAR-CANT is the most sensitive configuration to the fuel stored in the wing, with an importance factor of 28.9%, and the second most sensitive configuration is HAR SUGAR-SBW, with an importance factor of 18.1%. This can clarify the importance of this parameter for the WB masses of the HAR wings.

This study on the fuel mass stored in the wing might also provide an instinct for future hydrogen-powered aircraft configurations that either store the fuel in the wing-mounted drop tanks or fuselage and are likely not to have distributed fuel mass stored in the WB. This disadvantage may be offset by the shorter distance between wing spars, as demonstrated by the results in Figure 74. The trade-off between the distributed fuel stored in the wing, the distance between the spars, and the positions of the concentrated loads caused by the drop tanks is likely to raise a new design question and should be explored in future studies.

7.2.4.3 Wing Group Mass

The wing group mass is the total wing mass of an aircraft. As discussed in the previous sections, short computational times are intended in conceptual design-level studies. Non-iterative direct solutions from the model solvers reduce the computational time drastically compared to the iterative solutions. The comparison of the direct and convergent solver approaches, as previously discussed in Section - 0, and the direct approach's accuracy in wing box and wing group masses was stated to be enough. This study will further investigate the wing group mass effect with a parametric study.

The WB masses of all the configurations have the same trend against increasing wing group mass, as shown in Figure 79. The higher sensitivity of the WB masses of HAR wings to wing group mass compared to moderate AR wings can be seen in Table 29 and Figure 79. While the importance factors of SUGAR-SBW and SUGAR-CANT are 22.7% and 31%, respectively, Boeing-737 and Boeing-747 have importance factors of 5% and 5.5%.

Although the effect of wing group weight on moderate AR Boeing 747 and Boeing 737 is minimal, and the direct solver approach of the WB ME tool can be used, it has a higher effect on HAR SUGAR-SBW and SUGAR-CANT configurations, and an iterative solver approach might provide higher accuracy. The outcome of the parametric study of the wing group mass, therefore, outlines the importance of an accurate initial wing group ME

method for a direct solution approach and the sensitivity level of the WB mass to wing group mass.

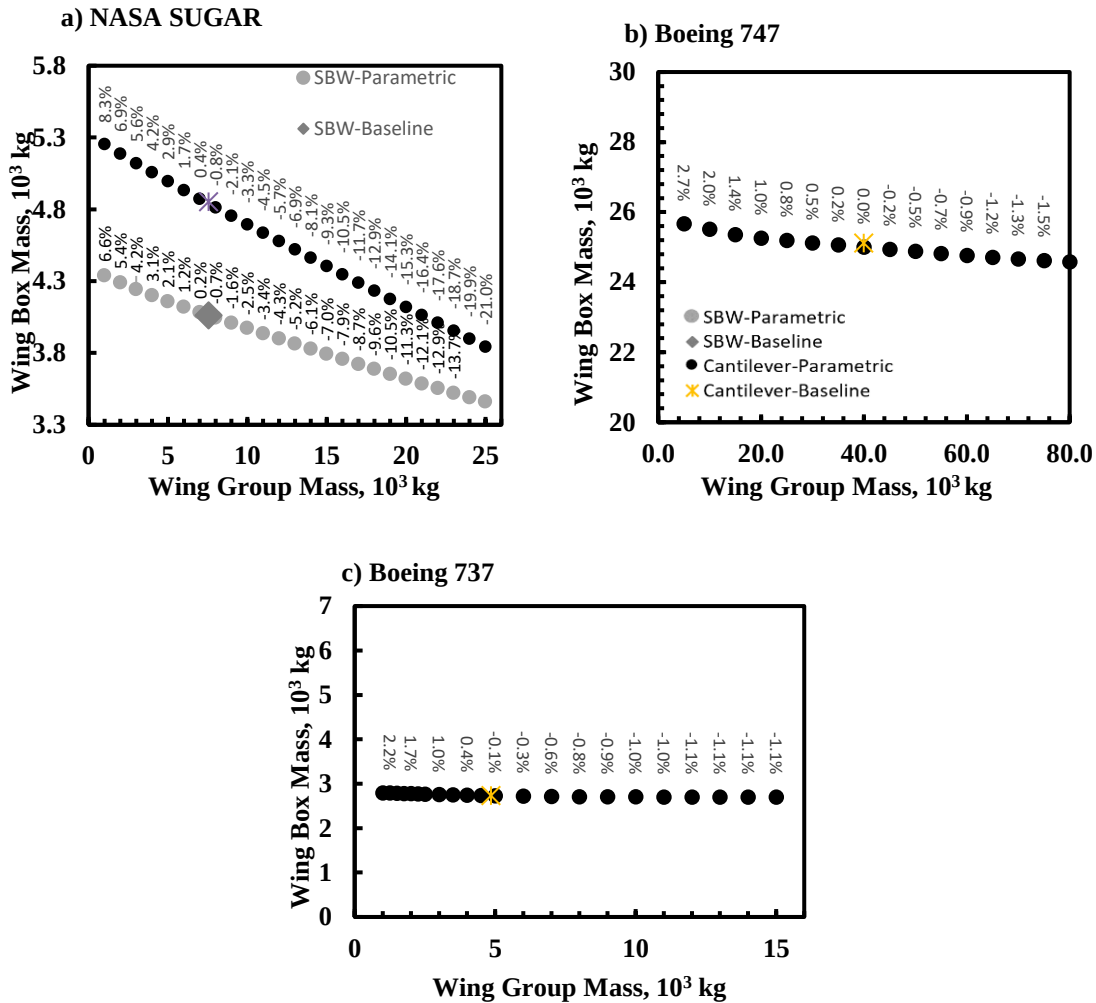


Figure 79 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737: baseline aircraft wing box mass changes with the change in the total wing group mass.

7.2.4.4 ND Spanwise Position of the Engine

ND spanwise position of the engine is a fraction of the wing half span length. As can be detected from Table 29 and Figure 80, the spanwise position of the engine has a greater impact on the WB weight of HAR wing configurations compared to the moderate AR wing of the Boeing 737 and 747. SUGAR-CANT has the highest importance factor of 37.9%, while the second most sensitive configuration is SUGAR-SBW, with an importance factor of 18.%. The importance factors for the Boeing 737 and Boeing 747 are 9.1% and 5.7%, respectively. The WB mass of HAR wings tends to reduce with more outboard engine

positions; however, the moderate AR wings do not show an obvious tendency compared to the HAR wing.

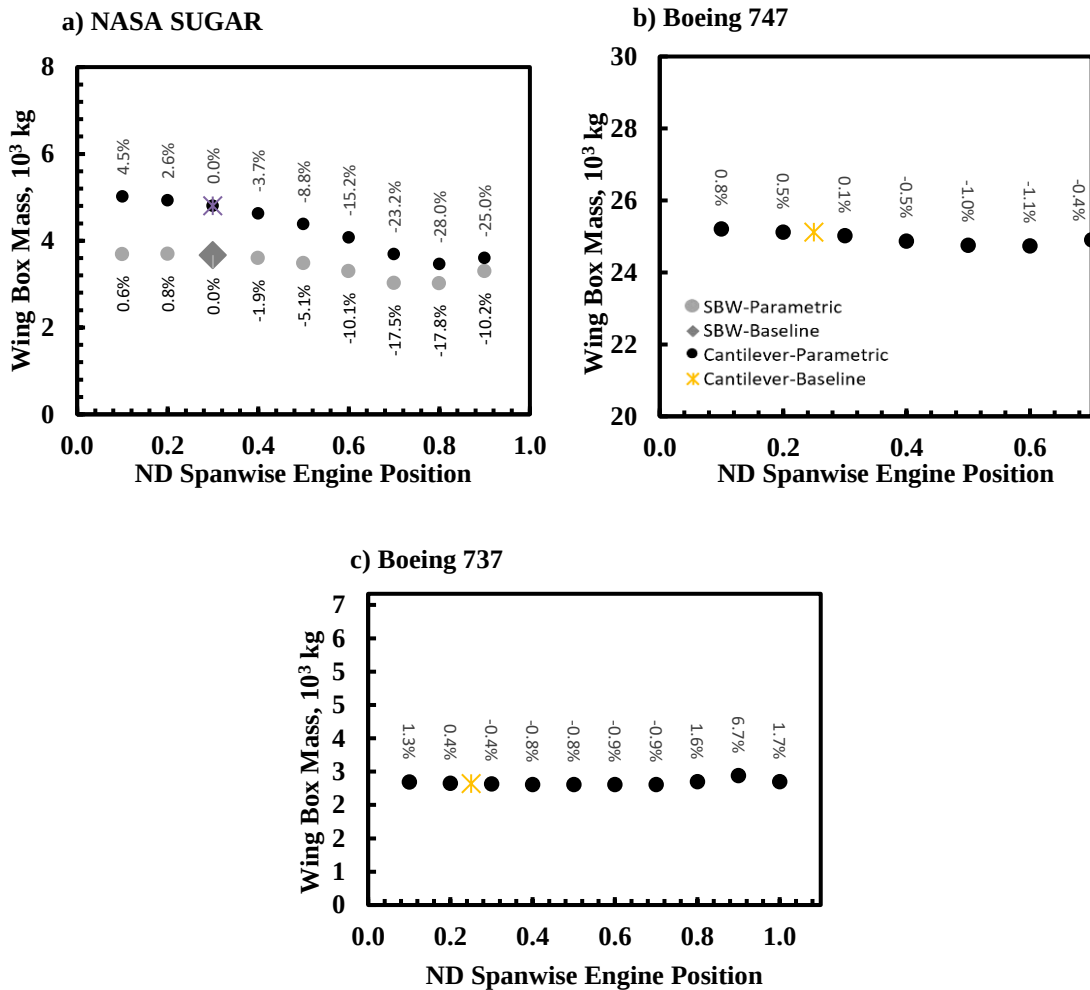


Figure 80 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737 baseline aircraft wing box mass change with the change of the ND spanwise position of the engine.

7.2.5 Wing Planform

7.2.5.1 Thickness to Chord Ratios of the Wing Root and Wing Tip

The thickness-to-chord ratios at the tip and the root of the wings are important parameters for all baselines; however, some configurations are more sensitive to those parameters than others. The sensitivity of the Boeing 747 and SUGAR-CANT to the thickness-to-chord ratio at the root is significantly higher compared to the others, with importance factors of 242.4% and 107.1%, respectively. The sensitivities of SUGAR-

SBW and Boeing-737 are measured with importance factors of 38.05 and 44.8%, respectively.

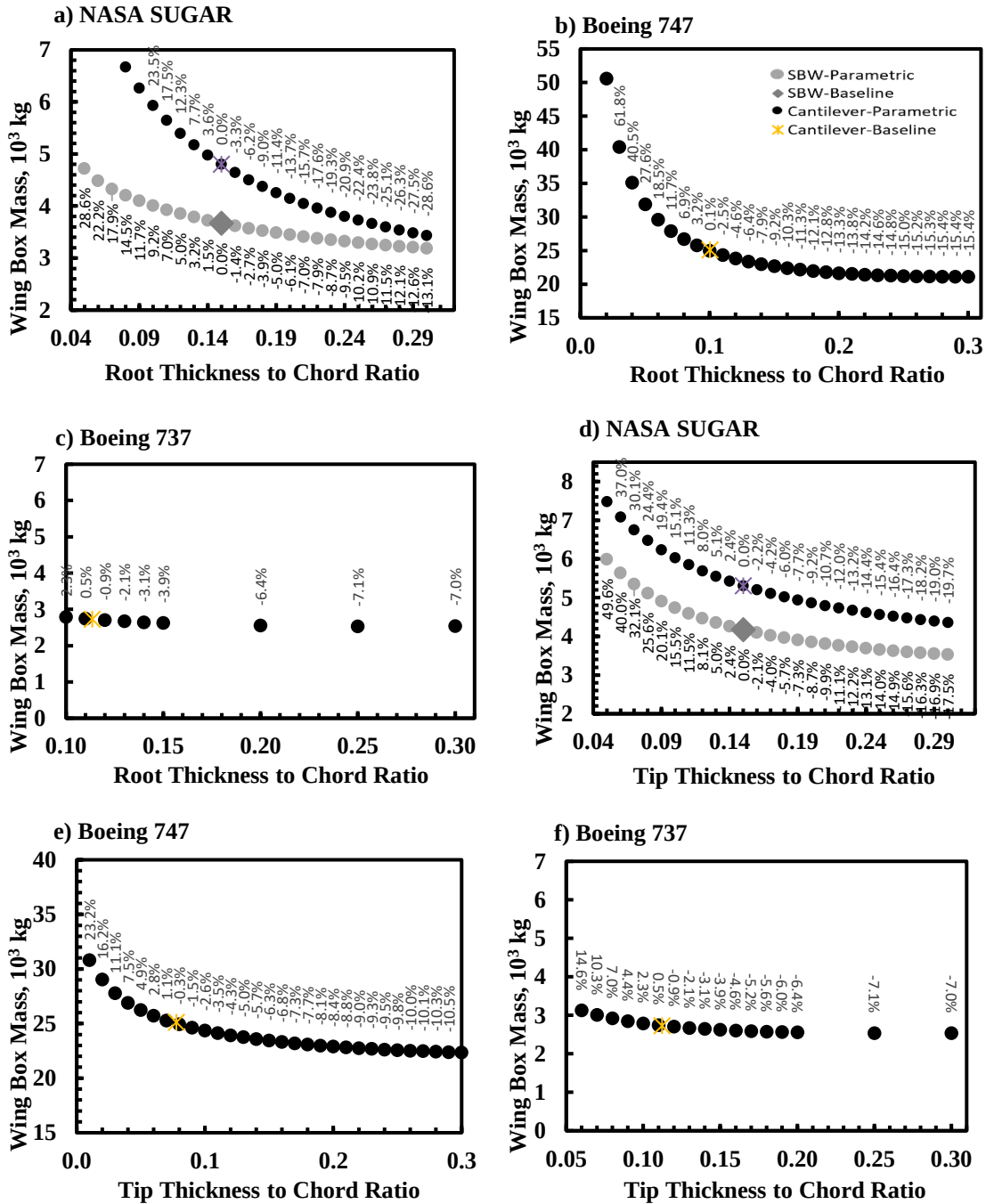


Figure 81 a) NASA SUGAR-SBW, SUGAR-CANT, b) Boeing 747, c) Boeing 737, baseline aircraft wing box mass changes as the thickness to chord ratio at the wing root changes. And d) NASA SUGAR-SBW, SUGAR-CANT, e) Boeing 747, f) Boeing 737 baseline aircraft wing box mass varies as the thickness to chord ratio at the wing tip changes.

It appears from Table 29 that HAR wings of SUGAR-SBW and SUGAR-CANT have more sensitivity to thickness to chord ratio at the tip, with importance ratios of 60.35% and 64.7%, respectively, compared to the importance factors of the Boeing 747 and Boeing 737, with 38.5% and 38.8%.

As shown in Figure 81, all configurations' WB masses tend to reduce with increasing thickness to chord ratios at the root and the tip. The trend of WB mass variation with the thickness-to-chord ratio value in this parametric study matches the result of previous studies from references (Olson and Henricks, 2018; Solano *et al.*, 2021; Sarojini *et al.*, 2022).

The reduced WB mass with the increasing thickness-to-chord ratio can be explained by the increased second moment of inertia of the wing box cross-section, which reduces the amount of bending-resistant material. With increased thickness to chord ratio, the distance between the stringers is also reduced, as this study's total stringer count is a fixed parameter. The reduced distance between stringers also reduces the thickness of the skin panels, reducing the required shear-resistant material.

7.2.5.2 Aspect Ratio

In this study, the wing taper ratio and wing surface area are kept fixed while the wing root, tip chords, and wingspan length are changed to change the AR.

The developed method seems sensitive enough to AR increases and the trend of the WB mass variation in graphs correlates with the results of the previous studies from References (Olson and Henricks, 2018; Solano *et al.*, 2021; Sarojini *et al.*, 2022).

Among the fourteen design parameters in Table 29, the wing box mass is the most sensitive to the wing AR for all the studied aircraft configurations. The wing box mass increases exponentially with the increasing AR for all the configurations, as presented in Figure 82. This increase can be explained by drastically increasing bending stresses on the load-carrying structures. As previously discussed in Section 7.2.3.1 (Stringer Counts) and other sections, bending-resistant material becomes dominant for HAR wings, affecting the importance factor of some design parameters like AR compared to moderate AR wings. The WB mass sensitivity to AR variation of SUGAR-SBW, SUGAR-CANT, Boeing 737, and Boeing 747 are measured with importance factors of 348.8%, 325.3%, 392.6%, and 274.5%, respectively.

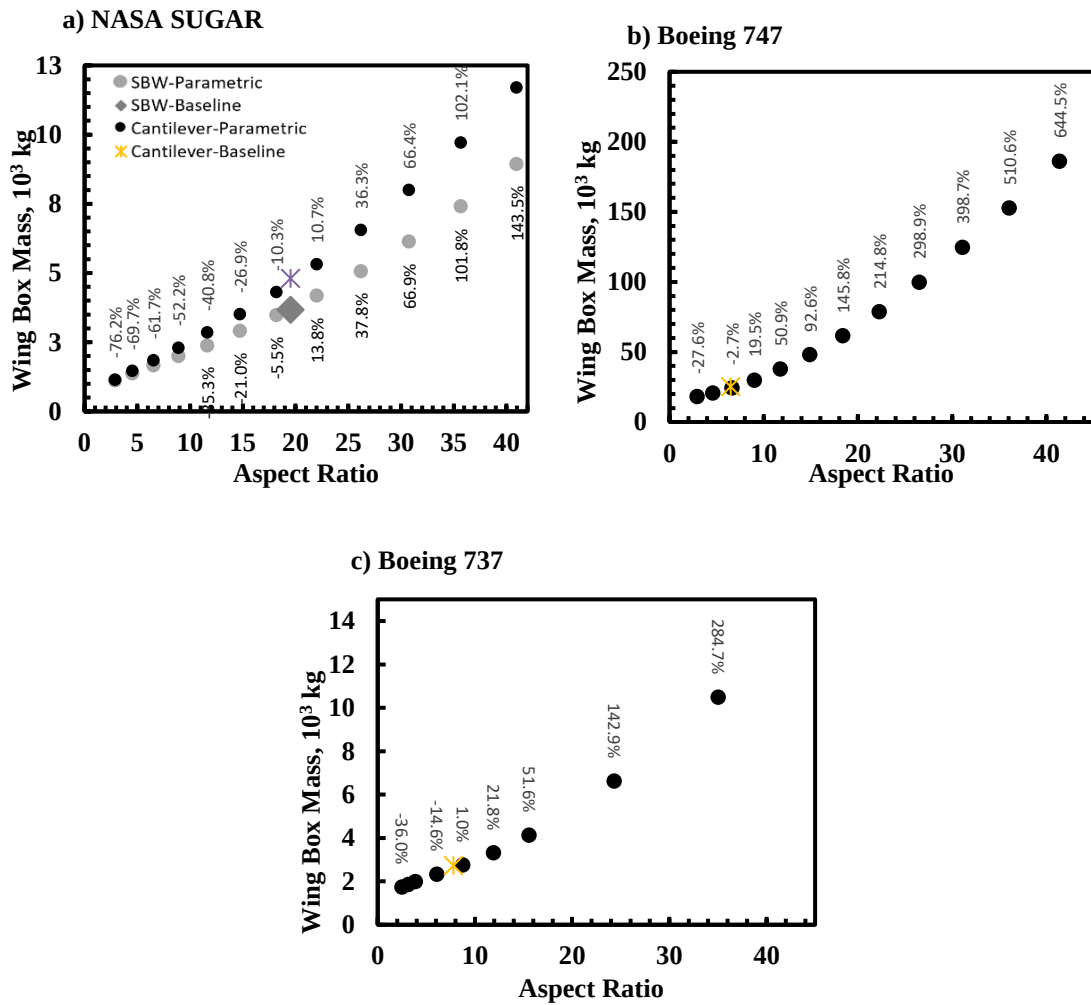


Figure 82 a) NASA SUGAR-SBW and SUGAR-CANT, b) Boeing 747 and c) Boeing 737, baseline aircraft wing box mass variation with wing aspect ratio change.

7.3 Concluding Remarks of Wing Box Parametric Study

After the validation studies, the effect of the strut on the WB weight reduction is investigated using quasi-isotropic composite and aluminium materials. It was demonstrated that the strut reduced the weight of the wing boxes by 11.9% and 16.2%, respectively, for the aluminium and composite wing boxes.

In the final section of this work, a comprehensive parametric study is conducted to investigate the effects of fourteen different aircraft design parameters on the WB mass of a HAR cantilever baseline of SUGAR-CANT, a HAR SBW baseline of SUGAR-SBW and moderate AR cantilever wing baselines of Boeing 747 and Boeing 737 using a one-factor-at-a-time sensitivity analysis approach. A simple importance factor from the

literature compares the effects of fourteen aircraft design parameters on the WB mass. The sensitivity of the WB mass to the studied parameters is also presented with detailed graphs generated from the variation of the WB mass value against the studied design parameters.

The optimum strut attachment point that achieves a minimum WB mass, excluding the structural mass of the strut, is found at an ND span length of 0.65.

The significance of the stringer count for four configurations is presented with importance factors and graphs; however, its trend was different for HAR and moderate AR wings. The increasing stringer counts increased the WB masses for HAR wings but decreased those for moderate AR wings. The mass increase in HAR wings is explained by the increasing dominance of bending resisting material in WB mass.

The effect of an idealized wing box shape on WB mass is also investigated. By changing the design parameter of the WB's shape, the idealized shape of the WB is gradually changed from a diamond shape to a hexagonal shape and a rectangular shape. The significant effect of an idealized WB shape is observed. For the HAR SUGAR-CANT configuration, the importance ratio of this parameter reached 90%. The WB mass decreases in all the configurations when the idealized WB shape approaches a rectangular shape.

The effect of the centre wing box length on the WB mass of SUGAR-SBW and Boeing-737 configurations was significant, with importance factors of 39.9% and 20.2%, respectively, compared to the other configurations. While increasing the centre wing box length caused a reduction in the WB mass of the SBW configuration, it increased the WB mass of the Boeing 737 baseline.

The effect of TOGW is higher for HAR SUGAR-SBW and SUGAR-CANT configurations compared to other configurations, with importance factors of 59.5% and 80.1%, respectively, even though the OEW importance values of all configurations have closer values. It's also observed that the WB masses diverge after they approach the minimum TOGW and OEW values, which might emphasize the importance of these two factors for an optimal wing box design. It was also demonstrated that increasing the mass of fuel stored in the wing box decreased the WB mass of all configurations due to the inertia relief provided. The significance of wing group weight on the WB mass is also shown for HAR wings. Because the wing-group mass is a dependent variable of the calculated WB

mass, iterative WB mass calculations for HAR wings are advised to converge to a final WB mass result.

It was shown that the WB mass is significantly sensitive to the thickness-to-chord ratio of the wing tip and root in all configurations. Increasing the thickness-to-chord ratio reduced the WB mass of all configurations, which correlates with previous research findings in the literature. The extreme effects of the wing AR on the WB mass of all configurations were also shown. An increase in the wing AR increased the WB mass of all configurations due to the increased bending stresses, and this trend also relates to the results of the previous studies in the literature.

In summary, the new WB ME approach is validated for HAR and moderate AR cantilever wings and HAR SBWs. In addition to its high accuracy, the method's sensitivity to aircraft design parameters has been successfully demonstrated. Important points from the parametric and comparative studies are highlighted to provide an instinct for conceptual design studies of novel and traditional aircraft configurations. The importance of covering the effects of fourteen aircraft design parameters to develop an accurate and sensitive enough conceptual or early preliminary design WB ME method is stated.

7.4 Truss-Braced Wing Detailed Parametric Study

The previous section investigates wing box mass variation with some design parameters of four baseline aircraft. In this section, a similar parametric study approach will be applied to investigate the effect of the design parameters of SUGAR-SBW on the masses of the wing box, wing group mass, strut, jury, and offset. The details of SUGAR-SBW are given in Table 57, and the baseline configuration is named SUGAR-TBW in this section to emphasize the jury's presence.

This section helps verify the developed new ME methods for TBW configurations. As the mass datasets of TBW are limited in the literature when this research is conducted, this verification study can help observe any possible errors or weaknesses of the new methods. Moreover, these studies also help increase the credibility of the complete mathematical model (MM) in general.

The details of the mass source of the wing group, wing box, strut, jury, and offset were discussed in Chapter 6 (MASS ESTIMATION) and presented clearly in Table 1. These sections can be reviewed to interpret better the results given in this section.

7.4.1 Strut Design Parameters

7.4.1.1 ND Spanwise Strut Position

In the previous section, the effect of the ND spanwise position of the strut was investigated by considering wing box mass only. This section studies its effects on all the main TBW components. As previously mentioned, the optimum strut position for the minimum wing box mass is around the ND spanwise position of 0.8 for SUGAR-TBW aircraft; however, strut mass increases exponentially with the more outer attachment of the strut, as shown in Figure 83. This mass increase is due to the growing strut length. Therefore, the total wing group mass should be considered to find the optimum strut position to achieve the minimum total mass. It can be seen in Figure 83 that the optimum strut attachment point is around 0.6 ND spanwise position for SUGAR-TBW aircraft. Although the jury has no clear pattern with changing strut position, it is affected, as shown in Figure 83.

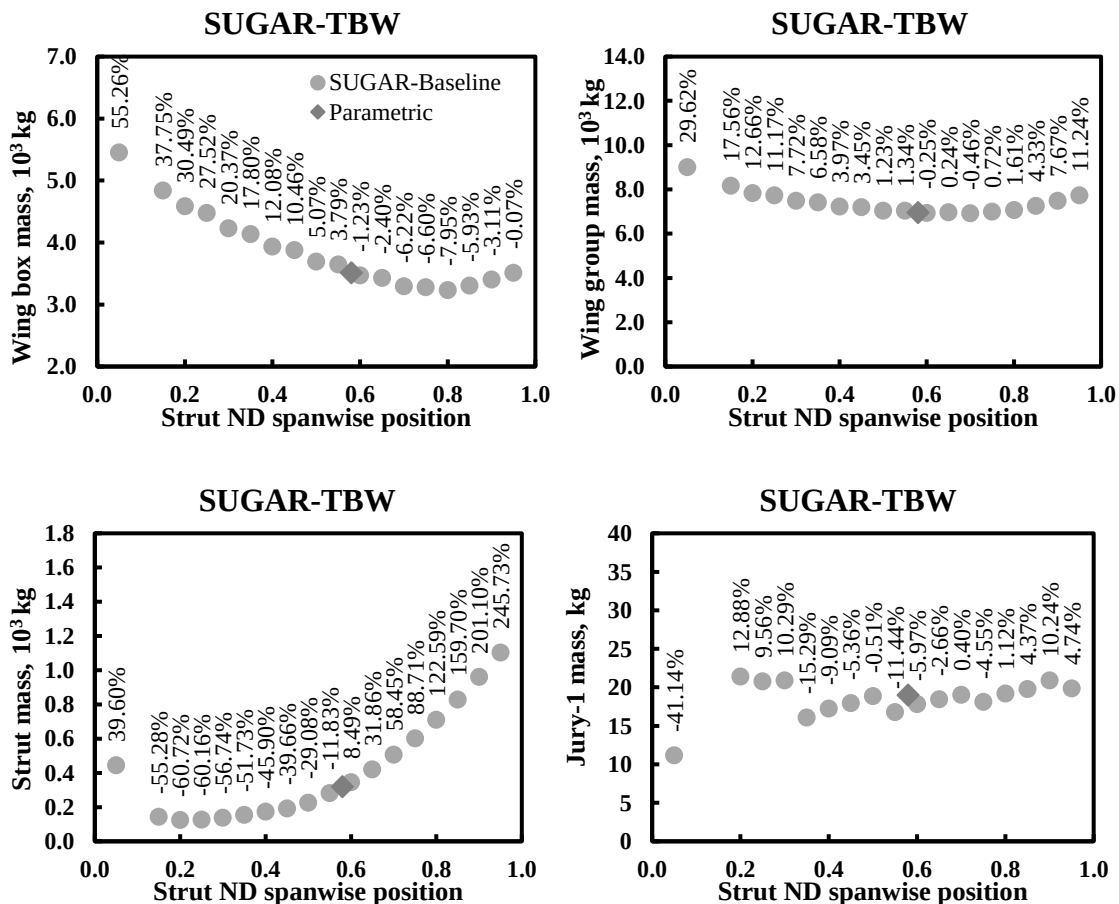


Figure 83 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the ND spanwise position change of the strut.

7.4.1.2 ND Chordwise Strut Position

Observations from Figure 84 highlight that the ideal strut attachment point for the minimum wing group mass is towards the leading edge, and alterations in the strut chordwise position do not influence the strut mass. This invariant behaviour of the strut mass can be attributed to the consistency of the strut length and the applied load, a consequence of the analysis method employed and the assumptions inherent in its development.

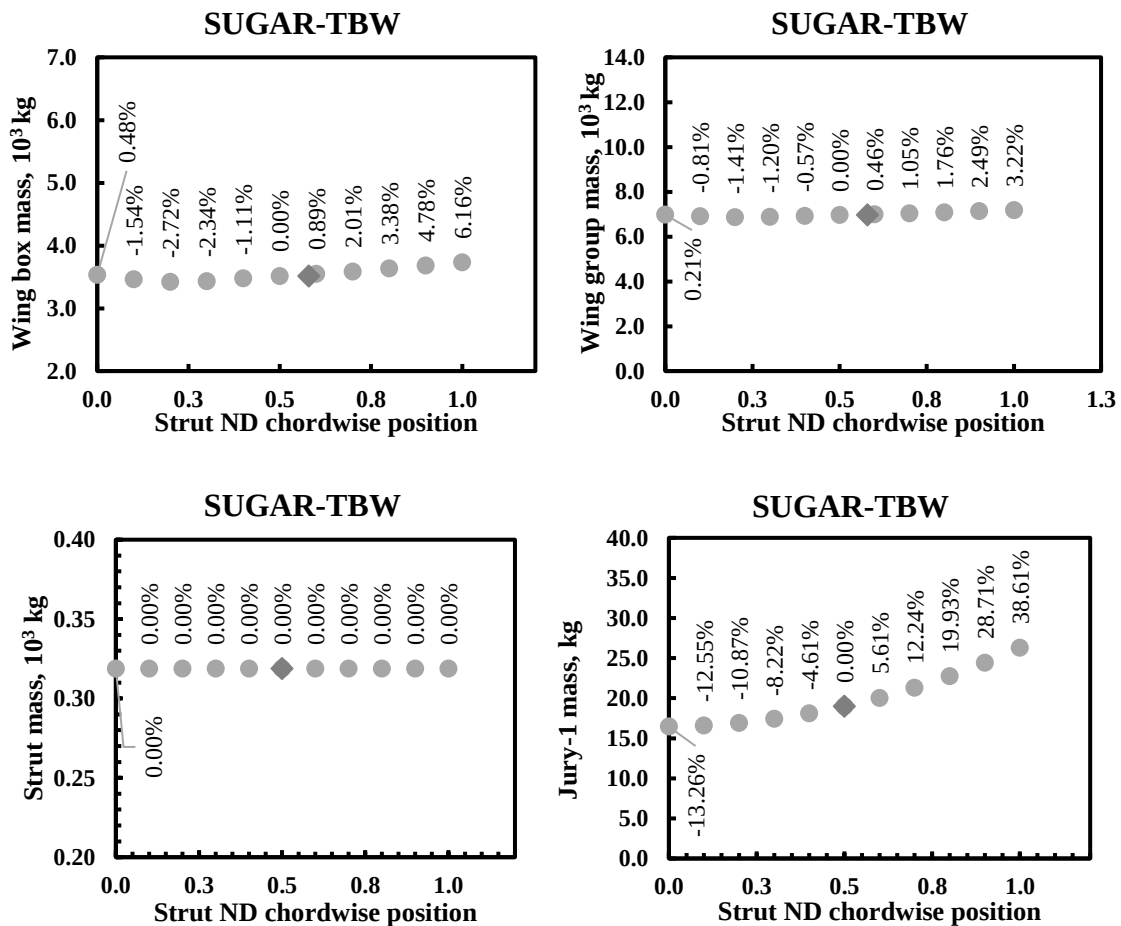


Figure 84 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the ND chordwise position chance of the strut.

Furthermore, the figure suggests that an approximate ND chordwise length of 0.28 serves as the optimal strut chordwise attachment point for minimizing both the wing group and wing box mass. As the attachment point approaches the rear spar, increased truss group mass is noticed, attributable to increased jury length.

7.4.1.3 Strut Sweep Angle

Figure 85 reveals an upward trend in the masses of all TBW components as the strut sweep angle increases. This escalation in truss group mass is attributable to the lengthening of the strut and jury. The wing box mass augmentation might be associated with amplified reaction forces induced by variations in the sweep angle.

Furthermore, the figure indicates that the strut sweep angle's optimal value is 0 degrees. Beyond a 5-degree sweep, the wing group mass increases exponentially, underscoring the significance of sweep angle control in TBW design.

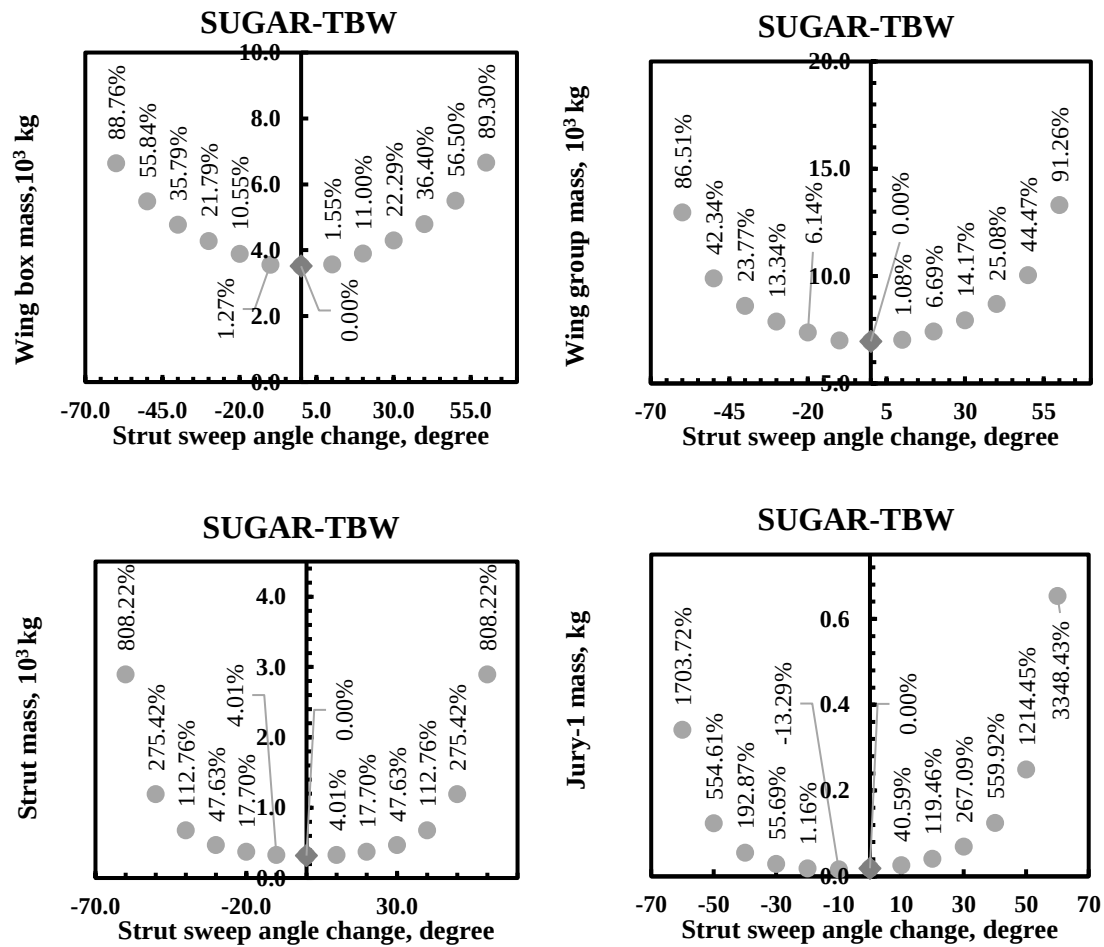


Figure 85 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the sweep angle change of the strut.

7.4.1.4 Strut Boundary Condition

The influence of the strut boundaries on the strut mass is due to the sizing of the load-carrying box of the strut for Euler column buckling criteria which is affected by the strut boundaries as explained in Section 5.5.1.1. It can be seen that only strut mass is

changed directly with the change of this parameter, and 21.90% lower strut mass is achieved when both ends of the strut are fixed.

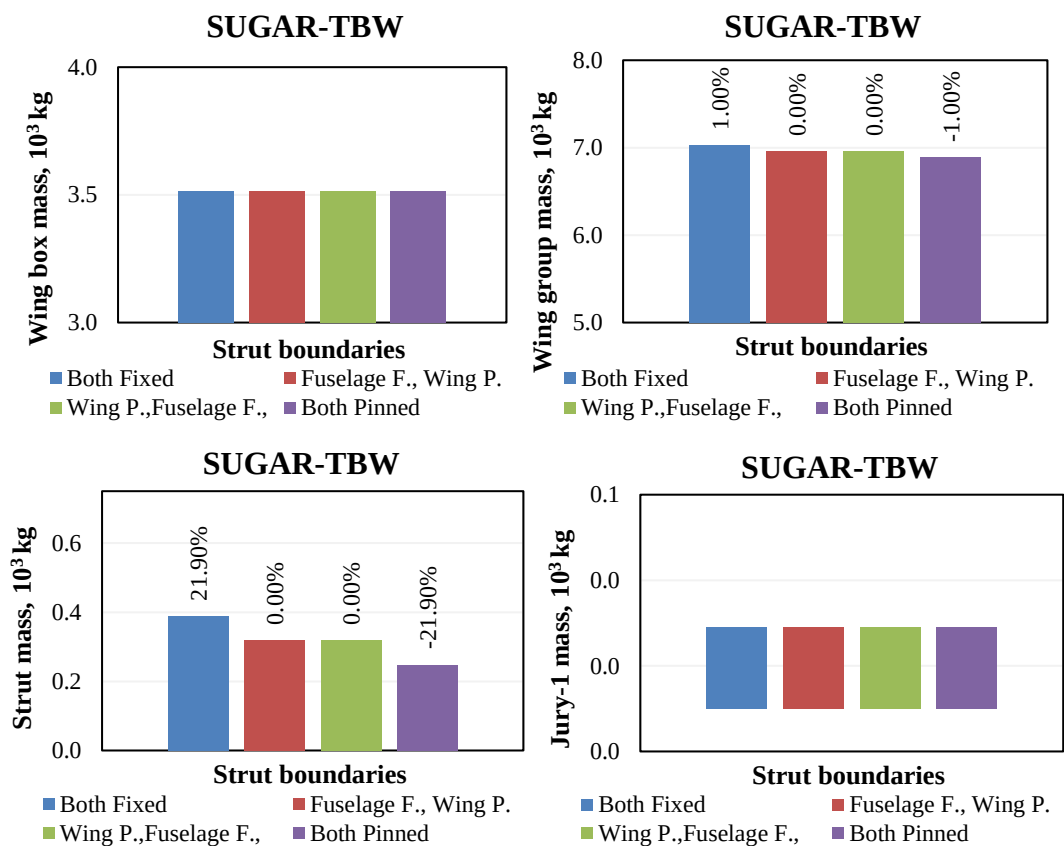


Figure 86 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the chance of the strut boundaries.

7.4.1.5 Strut Chord to Wing Chord Ratio

The parameter known as the strut chord to wing chord ratio is utilised to determine the value of the strut chord. Figure 87 shows that an augmentation in the strut chord beyond the optimal point of 0.4 for achieving the minimum wing group mass increases the overall wing group mass. This increase is primarily attributed to the amplified jury mass, which, in turn, is caused by the enlargement of the jury chord. However, it should be noted that the variation in the strut chord does not exert any discernible influence on the wing box mass within the scope of this model. This lack of impact may be attributed to the inherent simplifications and assumptions made during the development of the model, as previously discussed.

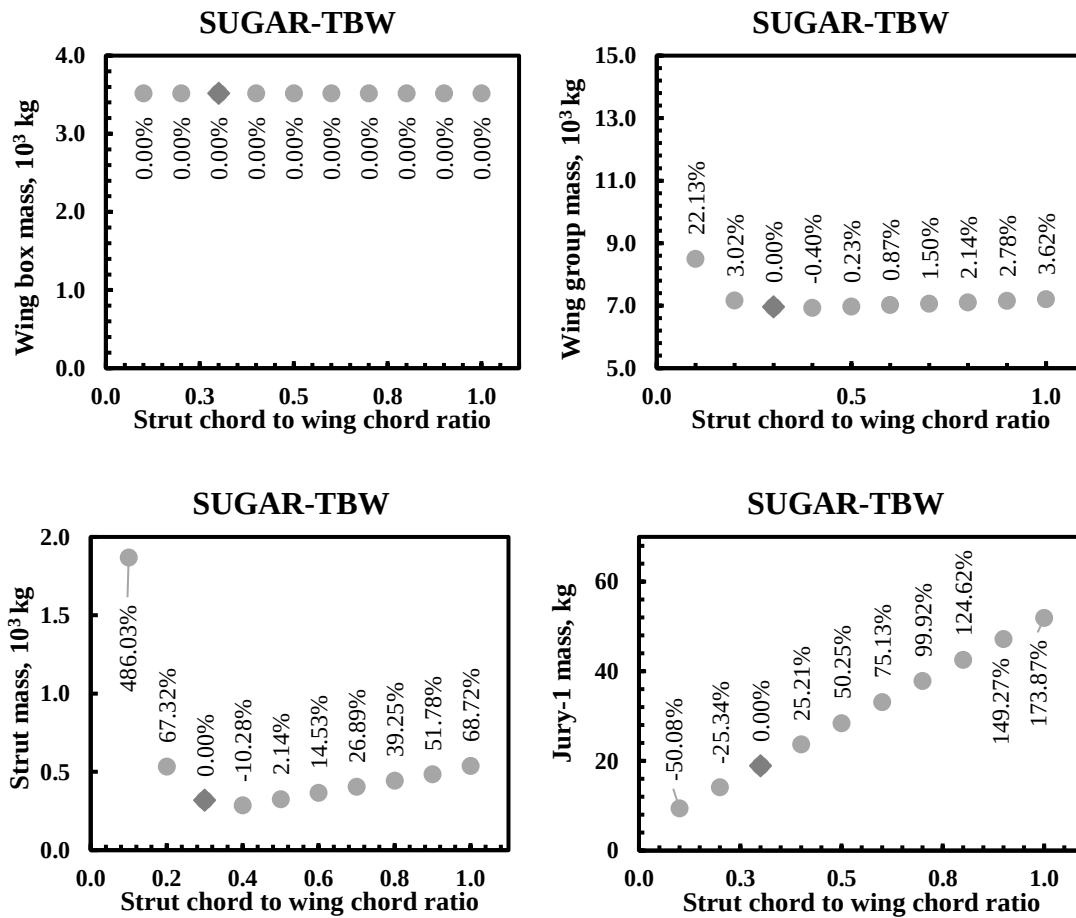


Figure 87 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the chance of the strut-to-wing chord ratio.

7.4.1.6 Strut Thickness to Chord Ratio

Based on the observations depicted in Figure 88, it becomes evident that as the ratio of strut thickness to chord surpasses 0.15, the mass of the struts stabilizes at a minimum value. Consequently, the truss and wing groups correlate with this pattern regarding their masses. Moreover, it is worth noting that the strut thickness to chord ratio does not exhibit any discernible effect on the masses of the wing box and jury components.

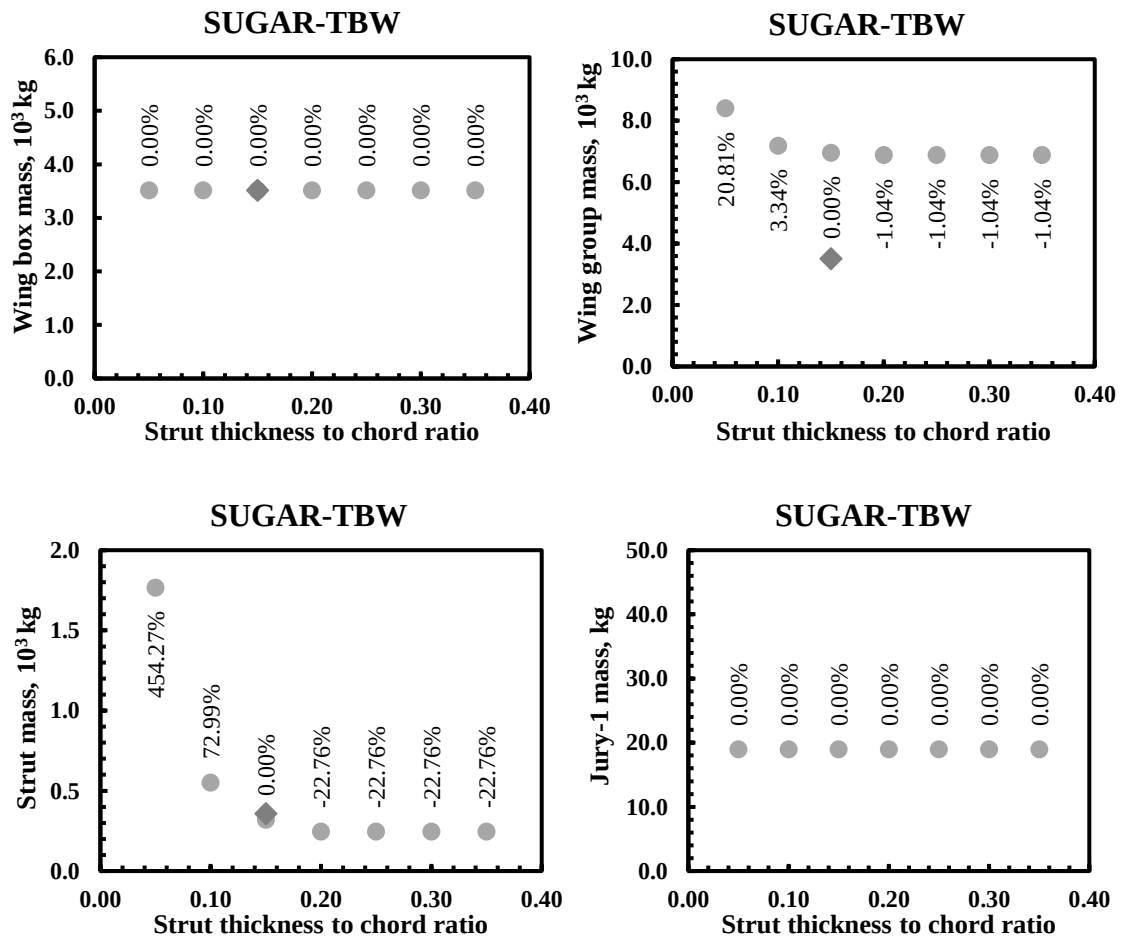


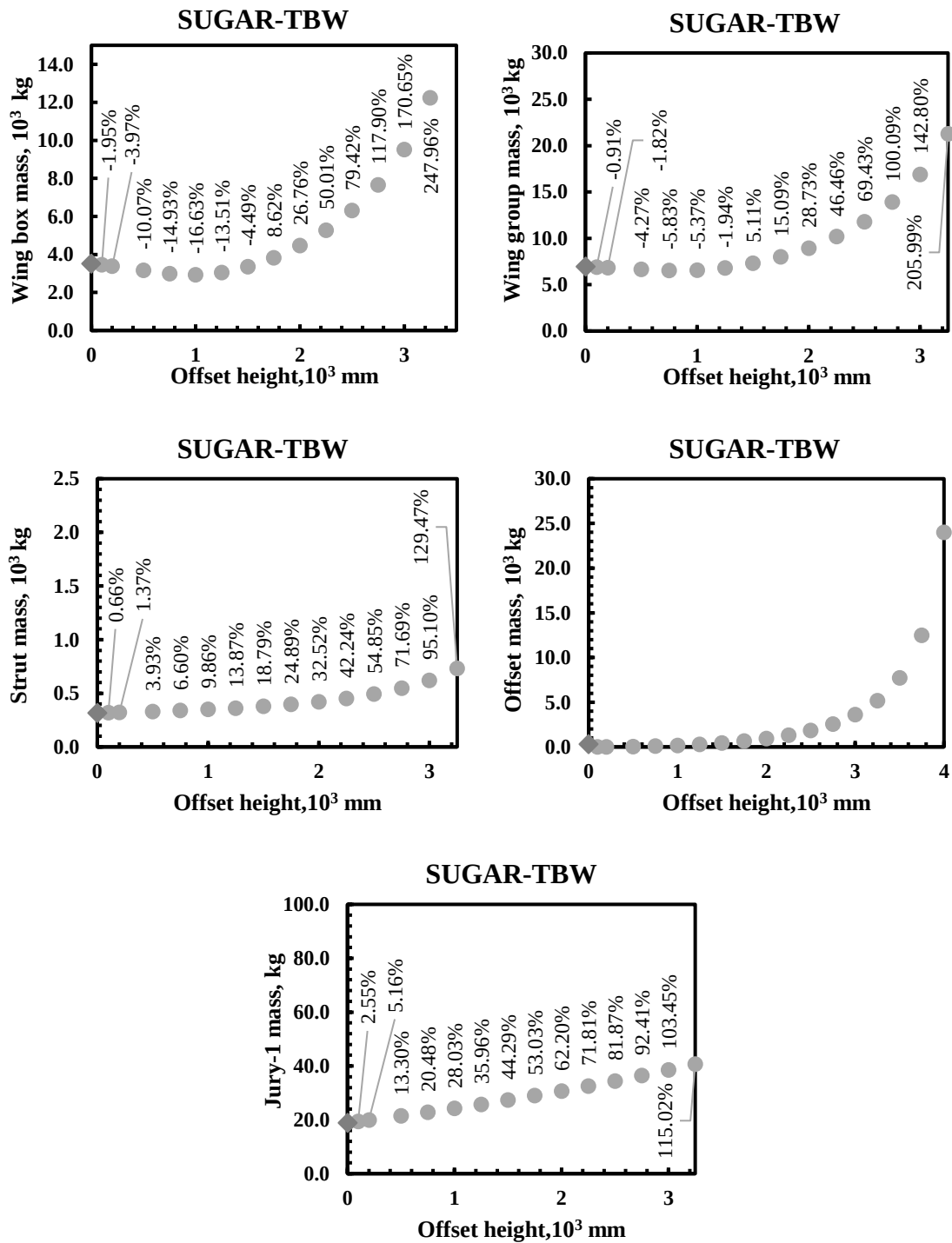
Figure 88 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the change of the thickness to chord ratio of the strut.

7.4.2 Offset Design Parameters

7.4.2.1 Offset Height

Figure 89 provides a crucial insight: incorporating an offset in the TBW configuration can lead to a local minimum in wing group mass due to the accompanying wing box mass reduction. For the SUGAR-TBW baseline, a significant reduction of 16.33% is observed when the offset size reaches 1m, concurrently marking the point of minimal wing group mass.

The increase in jury mass corresponds to the elongation of the jury with increasing offset length. Similarly, an elevated strut mass can be associated with changes in load paths within the jury, induced by increasing offset length. The exponential rise in offset mass with increasing length can be attributed to the escalating bending load at the offset root.



7.4.2.2 Offset Chord to Wing Chord Ratio

The offset chord-to-wing chord ratio is a defining parameter for an offset chord in the design process. As illustrated in Figure 90, an escalation in this ratio corresponds to a decrement in offset mass. The influence of this parameter on other components is consequential, as these components incorporate offset mass as a sub-component in their constitution.

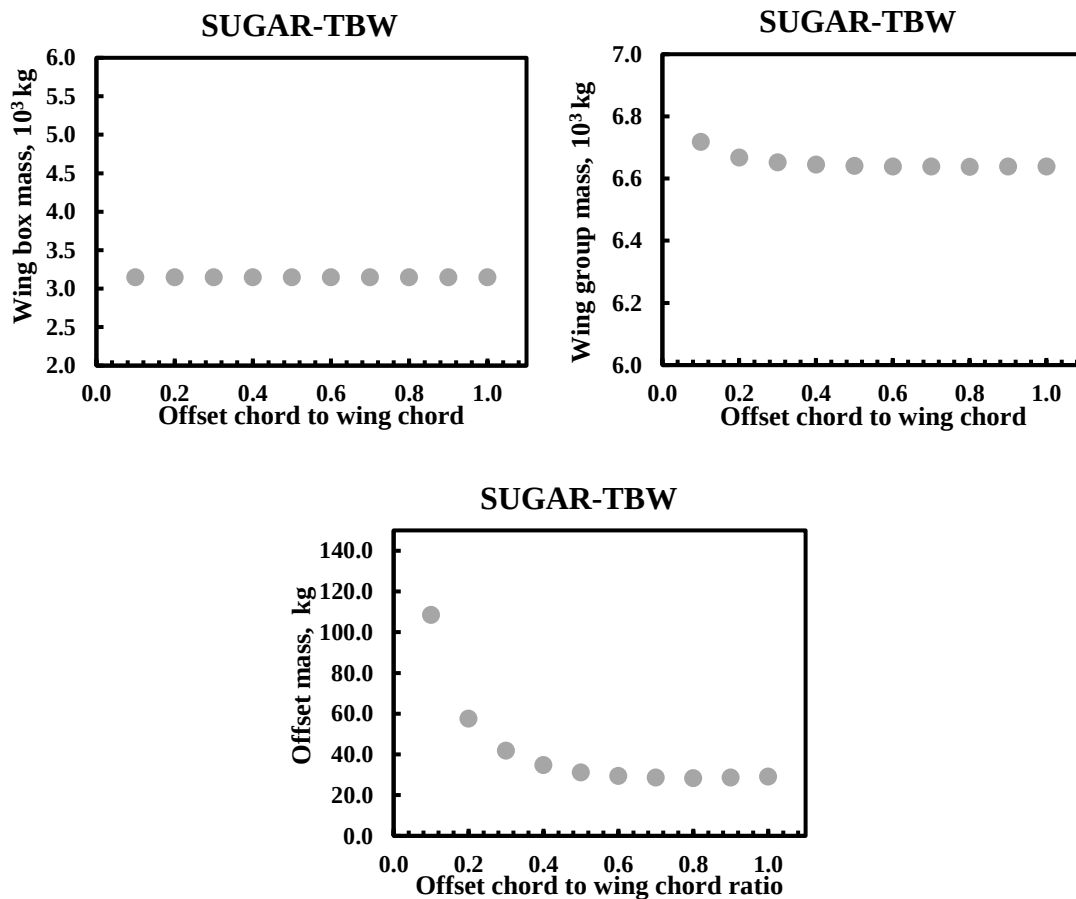


Figure 90 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the change of the offset chord to wing chord ratio.

7.4.2.3 Offset Thickness to Chord Ratio

An increase in the offset thickness to chord ratio increases the offset's thickness, which tends to lower the offset mass, as indicated in Figure 91.

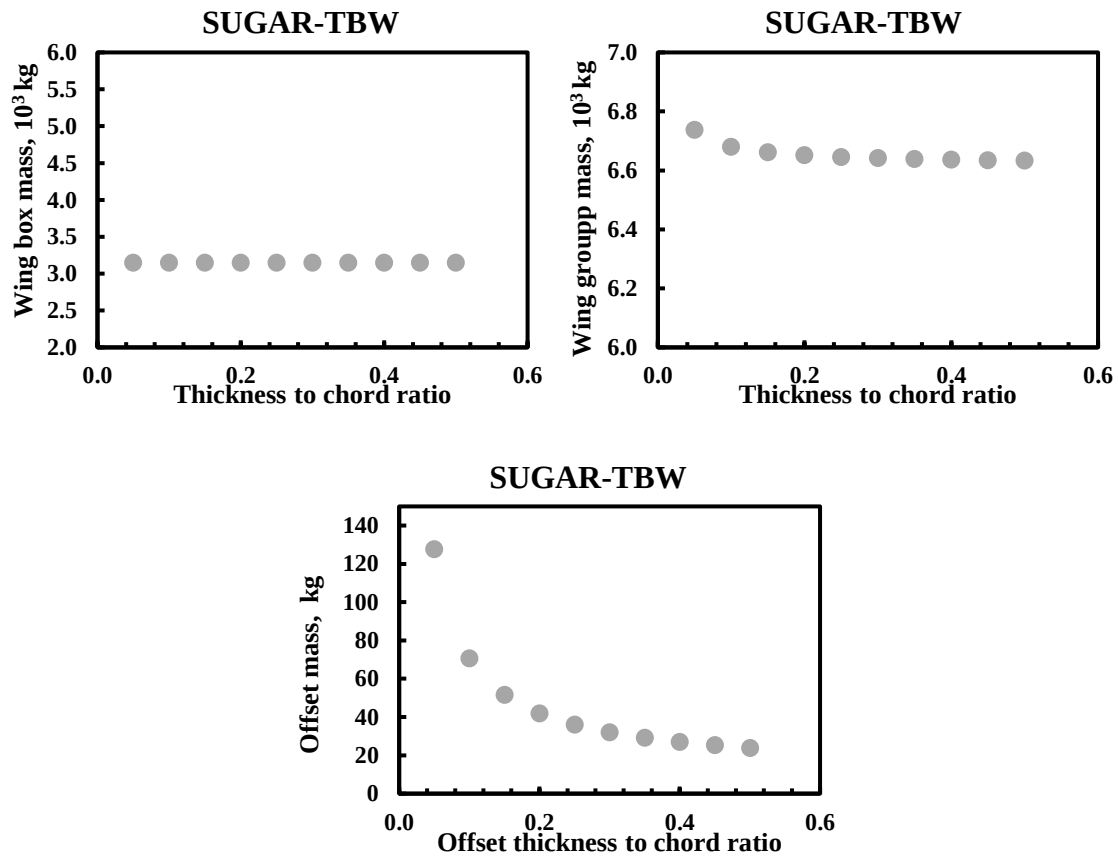


Figure 91 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the change of the offset thickness to chord ratio.

7.4.2.4 Offset Sweep Angle

A noteworthy effect of the offset sweep angle can be examined in Figure 92. While any rise or decline in this parameter increases the offset mass, the wing box mass reduces by 5%. A reduction in the wing group mass is also visible, around 2%. The mass growth and decrease in the jury are mainly related to the length of the jury. While those changes in the mass of the offset and wing box are likely to be caused by the load path change. These results indicate the consideration of this design parameter further.

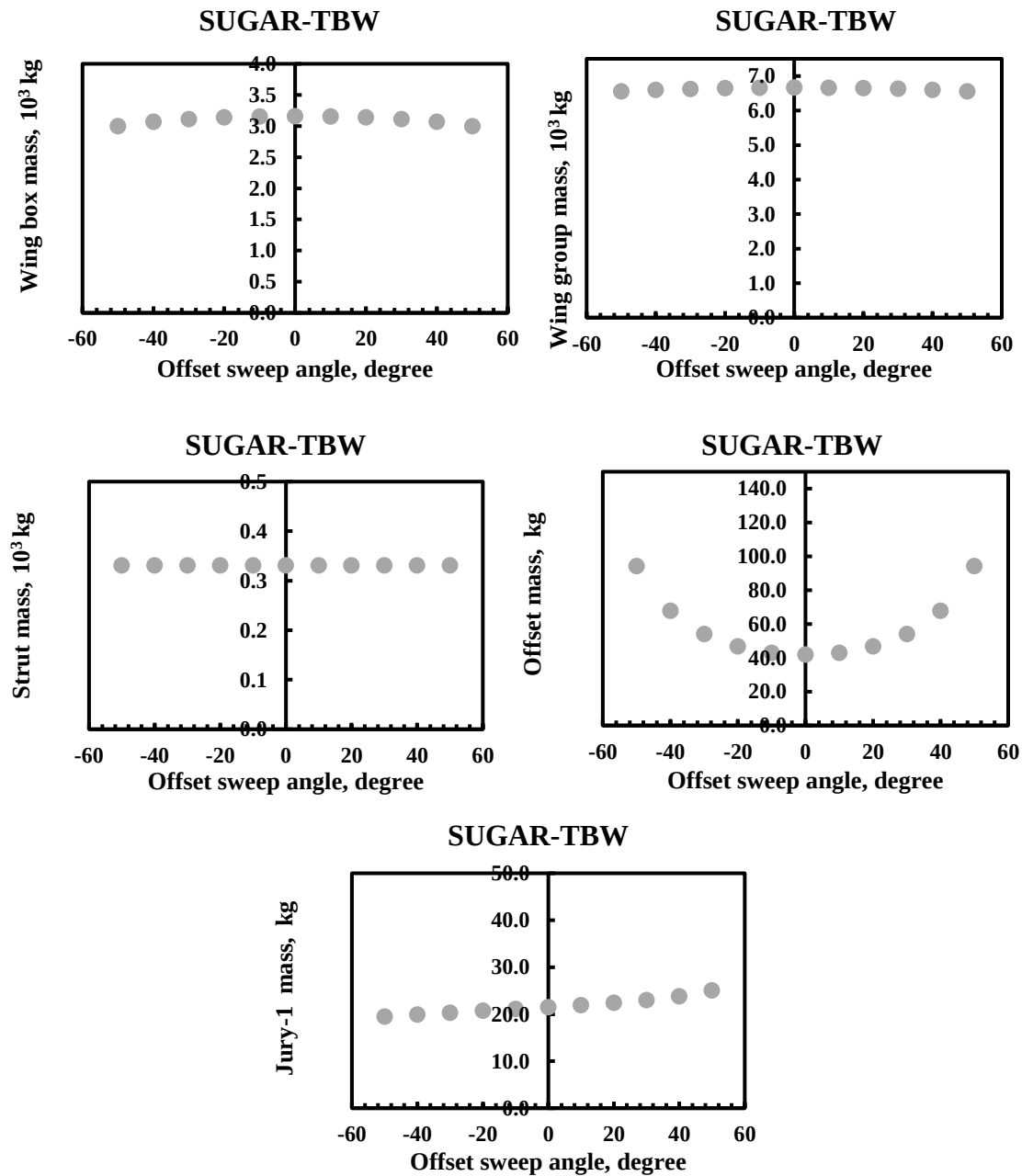


Figure 92 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the change of the offset sweep angle.

7.4.3 Jury Design Parameters

7.4.3.1 Jury Counts and Directions

In this analysis, the impact of jury direction and quantity on the mass of truss components was investigated, employing two jury orientations modelled in Section - 5.4: one perpendicular to the wing and the other to the strut.

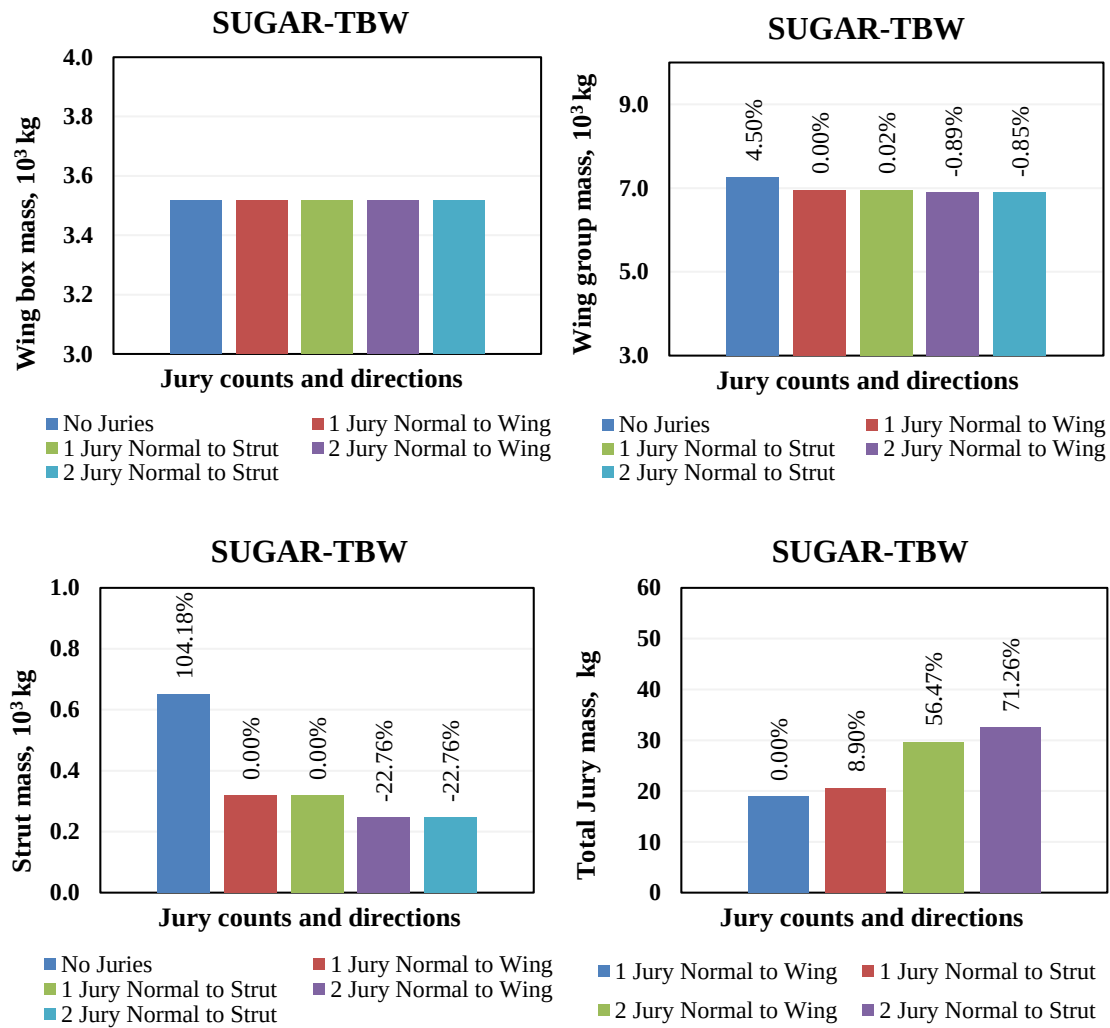


Figure 93 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the chance of the jury counts and directions.

As anticipated, it can be observed from Figure 93 that the juries contribute to diminishing the strut mass by mitigating strut column buckling. A strut without a jury yields a mass 104.185% greater than the baseline model with one jury. Further mass reduction in the strut (22.76%) occurs with the implementation of two juries. While this configuration increases the jury mass, the overall wing group mass diminishes by 0.89%. Nevertheless, introducing a second jury may induce heightened aerodynamic drag, posing limitations in multi-disciplinary studies. Importantly, the model indicates the wing box mass remains unaffected by the jury design.

Juries oriented perpendicular to the strut result in approximately 9% and 15% heavier weights for configurations with one and two juries, respectively, compared to their counterparts oriented normal to the wing. Although the jury direction has a marginal

effect on overall wing group mass reduction, it is advised to consider this factor during the preliminary design stage. Further, the influence of jury direction warrants additional examination in multi-disciplinary studies, particularly when assessing various TBW configurations.

7.4.3.2 Jury Chord to Strut Chord Ratio

The jury chord to strut chord ratio defines the jury chord. The increase in this parameter is examined to increase the jury mas constantly, as indicated in Figure 94. This predictable increase can be attributed to the enlarged LE and TE surfaces. While the strut and the wing box mass are not sensitive to this parameter, the wing group mass is affected inherently by the mass increase in the jury.

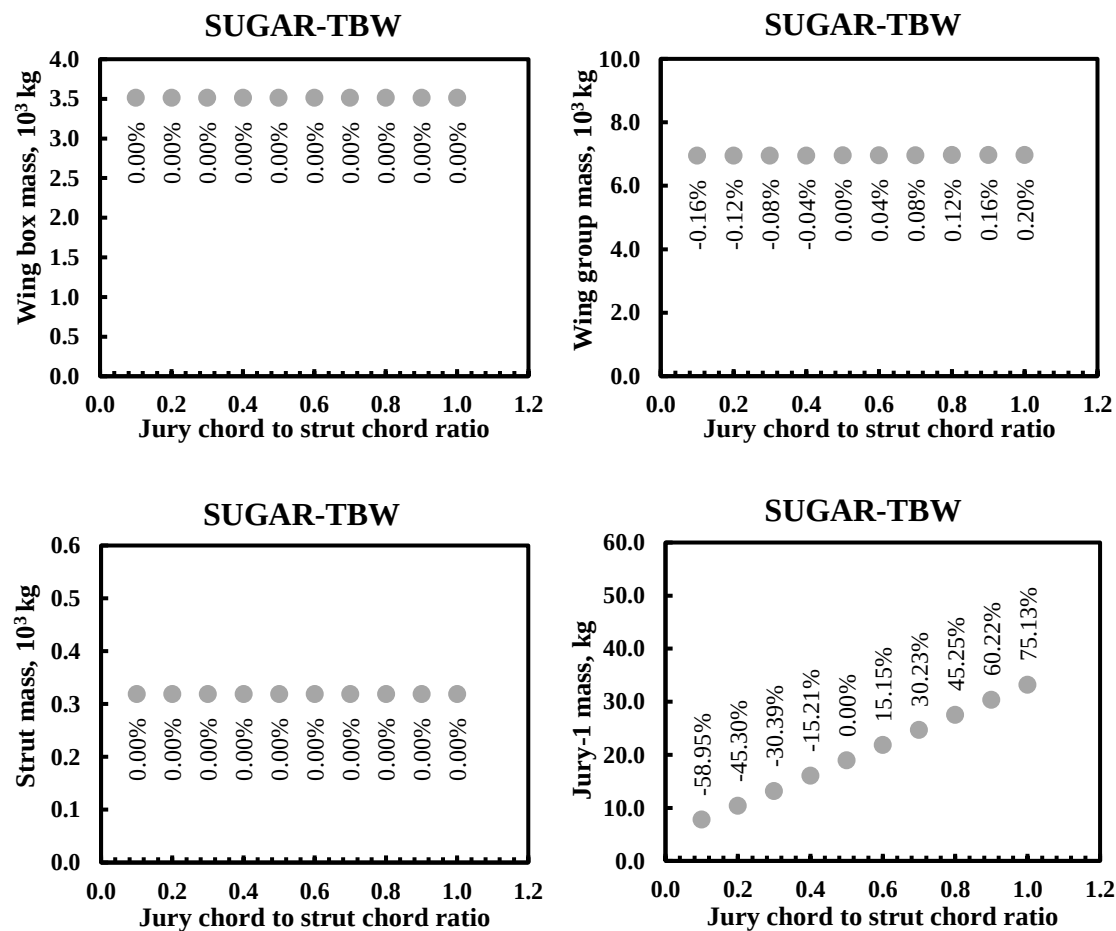


Figure 94 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the chance of the jury chord to strut chord ratio.

7.4.3.3 Jury Thickness to Chord Ratio

Figure 95 reveals that the rise in the jury thickness-to-chord ratio reduces the jury mass. This parameter is likely to be limited by aerodynamic constraints; hence it should be considered in a multi-disciplinary environment. While the strut and the wing box mass are unaffected by this parameter, the wing group masses are also reduced slightly with the increase in the tested parameter.

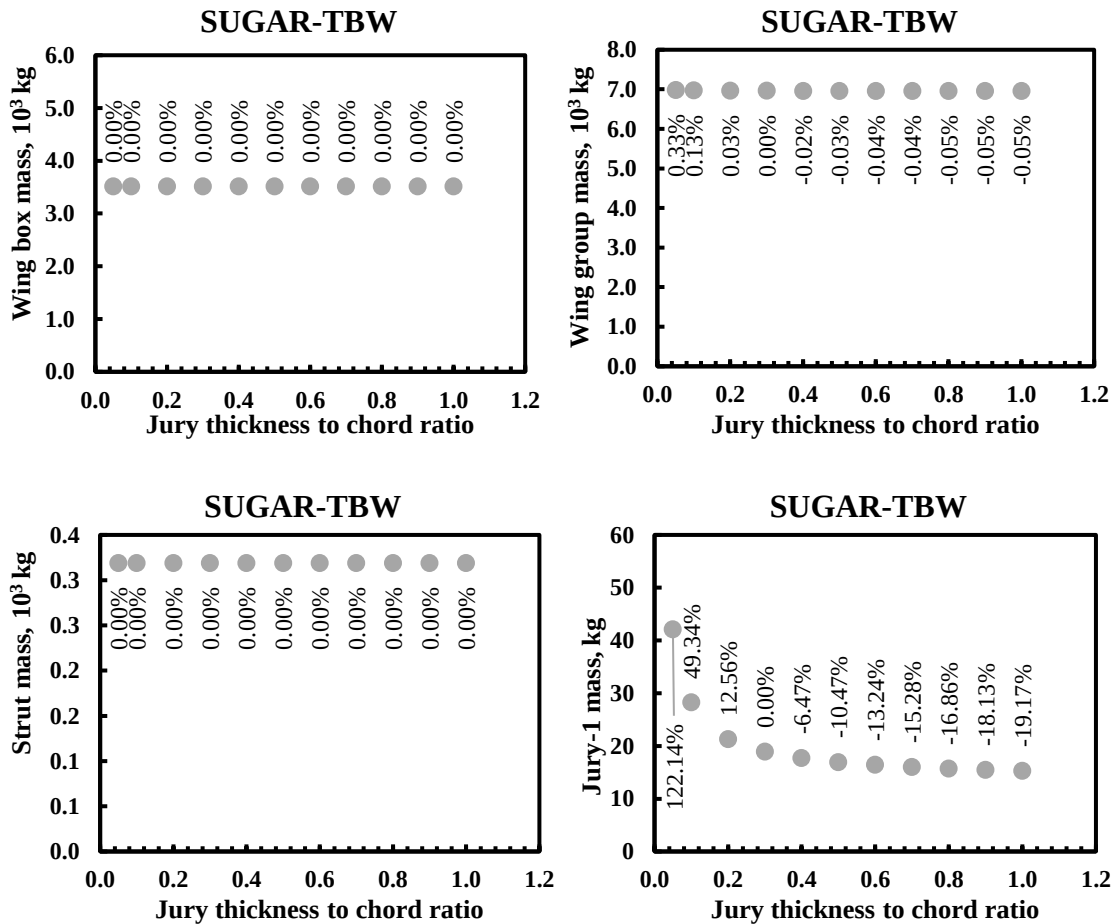


Figure 95 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the change of the jury thickness to chord ratio.

7.4.3.4 Jury-1 Location Change

The baseline aircraft featured a strut with boundaries fixed to the fuselage and hinged to the wing. These boundary conditions shift the optimal jury position for minimal strut mass from the strut's midpoint to 0.6 of the ND strut length, as depicted in Figure 96. As the jury position nears the wing-strut junction, the jury mass decreases due to the shortening of the jury length. Under these specific strut boundary conditions, the optimal jury attachment point for the lowest wing group mass is 0.6 ND strut length.

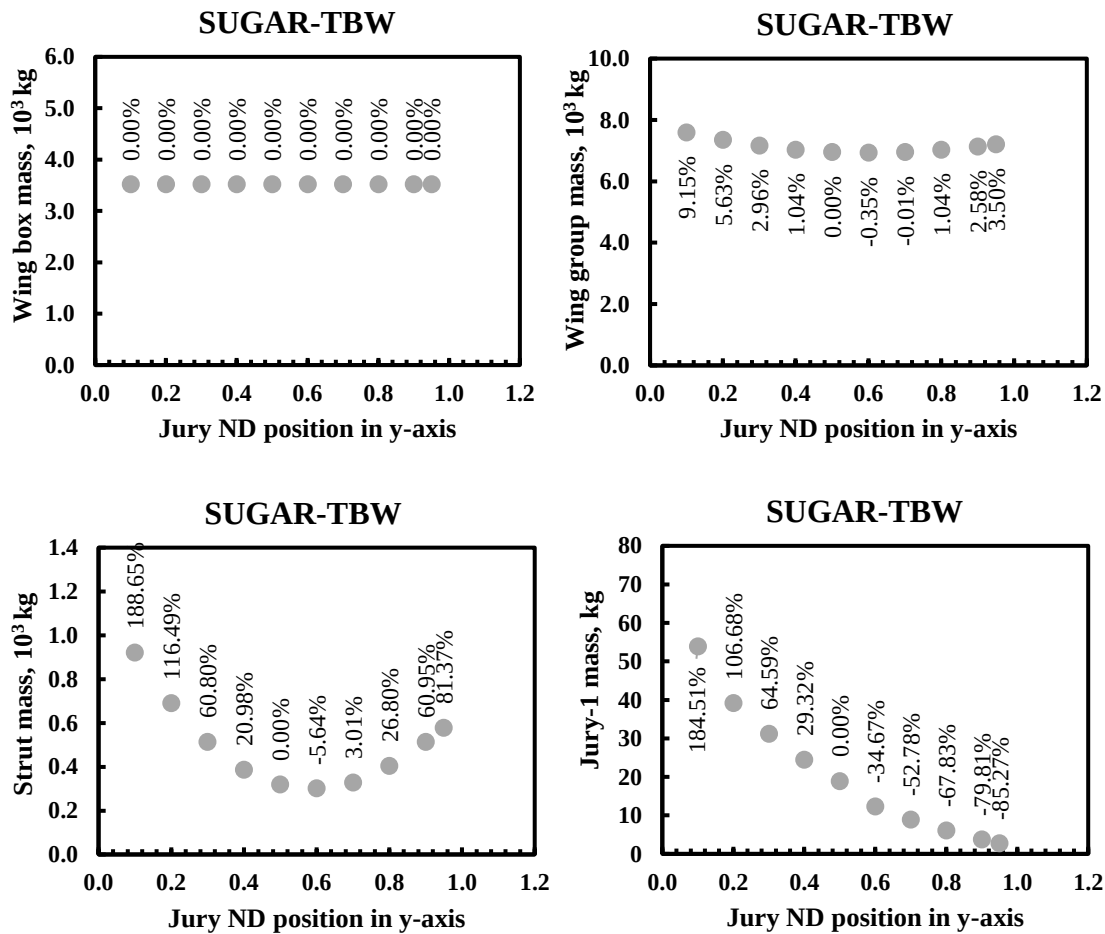


Figure 96 The mass variations of the wing group, wing box, strut, jury, and offset of SUGAR-TBW aircraft with the chance of the jury ND position in the y-axis.

7.5 Concluding Remarks of TBW Parametric Study

The exhaustive parametric studies conducted in this research offered significant insights into the impact of various design parameters on the structural masses of the TBW configuration, specifically focusing on the SUGAR-SBW aircraft. This research has deepened the understanding of these parameters and their interdependencies in SSaME, thus paving the way for improved designs and enhanced performance.

The optimal parameters were found to vary for different components of the TBW configuration, necessitating a holistic design approach considering the overall wing group mass. For example, the study discovered that the optimal ND spanwise strut position was 0.8 for minimal wing box mass, while it was 0.6 when considering total wing group mass, which includes the strut mass. The optimal spanwise position was determined for the SUGAR-SBW aircraft, wherein the strut is uniquely positioned, not at

the centerline, but at a distance of 3500mm, aligned with the landing gear pylon. Similarly, the ND chordwise strut position notably affected the wing box and truss group mass.

The study has also shown the importance of geometric parameters such as strut sweep angle, boundary conditions, and strut chord-to-wing chord ratio. The strut sweep angle of 0 degrees was optimal, indicating that variations in sweep angle can substantially affect the strut and jury lengths, thus influencing their masses. A significant reduction in strut mass was achieved when both ends of the strut were fixed, highlighting the influence of strut boundary conditions. The strut chord-to-wing chord ratio was found to have an optimal point at 0.4 for minimum wing group mass, beyond which the overall mass increases.

The inclusion of an offset in the TBW configuration revealed surprising findings. Remarkably, an offset can induce a local minimum in wing group mass due to the associated reduction in wing box mass, contrary to initial expectations. This discovery emphasizes the need for more in-depth offset design and optimization studies.

Lastly, the jury design parameters such as jury direction and quantity and jury chord-to-strut chord ratio have significantly influenced the mass of truss components. The presence of juries was found to mitigate strut column buckling, reducing strut mass.

This research has provided invaluable insights into the design of TBW configurations. The parametric study has substantially contributed to the current understanding of TBW design and has helped refine the developed new ME methods for TBW configurations. However, the findings suggest that a multi-disciplinary approach is necessary to account for factors such as aerodynamic drag and accurately determine the optimal parameters for different TBW configurations. Further research is required to refine these results, potentially in combination with other disciplines and higher-fidelity models.

8 CRITICAL LOADING CASES INVESTIGATION

This study will thoroughly explore critical loading cases for novel aircraft wing configurations such as short to ultra HAR cantilever, strut- and TBWs with distributed propulsion and hydrogen, electric, or fossil fuel technologies. The goal is to highlight the crucial loading scenarios, improving estimation accuracy and drastically reducing computing time and effort.

Accurate SSaME is pivotal in evaluating these new wing configurations. However, their inherent complexity often calls for numerous loading cases, escalating computational effort. Therefore, discerning the critical loading cases that steer SSaME becomes essential for an efficient and accurate evaluation of these novel concepts.

Addressing a gap in the literature, our study offers a comprehensive analysis of the dominant loading cases impacting structural sizing in novel SBW and TBW, strut, jury, and offset boxes. The developed mathematical model (MM) involving 483 loading cases has been utilized, analysing them to identify the most crucial ones for structural sizing.

This research serves two primary purposes:

- It is aimed to decrease the total computational time significantly by reducing the number of structural cases, a critical consideration when running optimization studies. The final optimised layout can then be verified by running the model with the full loading cases.
- Highlighting the most important loading cases can guide future researchers to focus on these and avoid errors caused by neglecting them.

The following sections will explain the methodology for identifying critical loading cases and the implications of our findings for SSaME of novel aircraft wing configurations.

8.1 Methodology

This section delineates the methodology to inspect the critical loading cases relevant to the SDaME of novel aircraft wings. The methodological framework encompasses the following stages:

- **Model Utilisation:** The MM with full loading cases, previously developed and elucidated in this thesis, is deployed for the SDaME of novel aircraft wings.

- **Data Generation:** The Latin Hypercube method generates a dataset comprising 1941 aircraft design samples. Each sample is characterised by approximately 71 input variables to attain the wing box and wing group masses being the output.
- **Model Application:** The MM is applied to these design samples, considering 483 loading cases for each aircraft. Every wing box is sectioned into 50 parts; each segment is sized based on the worst loading case. Thereby, 197,367 wing box structural elements, including skin panels, spar webs/caps, and stringers, are sized. The worst loading case drove the final size of each component.
- **Data Recording:** The sizing results for each wing section, and its corresponding worst loading case are meticulously recorded.
- **Data Analysis:** The critical loading cases are analysed by organizing the recorded data into a pivot chart in Excel, facilitating the identification of the loading cases contributing the most to the sizing of wing components and sections.
- **Case Identification:** Critical loading cases significantly impacting component sizing are identified and categorized based on their influence, with some cases affecting more components and sections than others.

This methodology aims to yield insightful revelations regarding the dominant factors governing the structural sizing of novel aircraft wing configurations, thereby enriching the overall comprehension of the design process and streamlining the optimization of the MM for SDaME.

8.2 Sampling Methods

Regarding the study presented in this thesis, sampling methods can be used to generate the datasets from the MMs for regression analysis or artificial intelligent models as surrogate models of the MMs. In addition to that, in this study, sampling methods are utilised to generate a dataset for critical loading case analysis. Several sampling techniques are available, each with its advantages and drawbacks. This section briefly discusses the most common sampling methods and focuses on the Latin Hypercube Sampling (LHS) method, its applications in similar research, and why it is suitable for the present study.

8.2.1 Common Sampling Methods

Random Sampling: A straightforward method where samples are chosen randomly from the input space. However, this method can lead to an uneven distribution of samples and requires a larger sample size to achieve adequate coverage (McKay, Beckman and Conover, 2000)

Stratified Sampling: This method divides the input space into strata and selects samples from each stratum to ensure uniform coverage. It can provide better representation and more accurate results than random sampling but may require more complex calculations (Cochran, 1977).

Sobol Sequences: A quasi-random sampling method that uses low-discrepancy sequences to achieve a more uniform distribution of samples. Sobol sequences can result in better coverage of the input space but may require more sophisticated algorithms (M., 1976)

8.2.2 Latin Hypercube Sampling (LHS)

Latin Hypercube Sampling is a stratified sampling method that divides each input variable into equal intervals and selects one sample from each interval (McKay, Beckman and Conover, 2000). LHS ensures a more uniform distribution of samples across the entire input space, which can improve the accuracy and efficiency of the analysis.

LHS has been applied in various aerospace engineering studies, including those focusing on ME and the structural design of TBWs (Chiozzotto, 2017). The method has proven effective in providing a balanced representation of input variables while minimizing the required sample size.

8.2.3 Suitability of LHS for the Present Study

The use of LHS in this study is appropriate for several reasons:

Uniform Coverage: LHS provides a uniform distribution of samples, ensuring that all areas of the input space are adequately represented.

Efficiency: The method requires a smaller sample size than random sampling, reducing computational costs and time.

Relevance: LHS has been successfully applied in similar research on aircraft wing design, indicating its suitability for the present study.

In conclusion, Latin Hypercube Sampling is suitable for generating samples from the MM used in this research. Its ability to provide uniform coverage, efficiency, and relevance to similar studies makes it an appropriate choice for analysing critical loading cases and training AI models in the context of novel aircraft wing design.

8.3 Results

71 aircraft design variables are manipulated using the Latin Hypercube method to generate the input variables of 1941 aircraft containing short AR to UHARWs, cantilever wings, SBWs and TBWs with conventional and distributed propulsions using electric, hydrogen or fossil fuel. As a result, 197,367 wing components are sized following the sizing methods explained in Chapter - 5 (STRUCTURAL ANALYSIS AND SIZING). The worst loading case defined each component's final size. Considering that the total component count sized by each loading case was recorded, this data was processed using a pivot chart in Excel. The loading cases are filtered to remove the loading cases which size fewer components than the filtering criteria.

Table 30 listed nine loading cases which sized more than 5000 wing components except for Case-3, which is critical for strut sizing of TBW and added in the critical cases. The loading cases, which sized more than 2500 wing components, are listed in Table 31. Moreover, similar to

Table 30 and Table 31, other tables (Table 62, Table 63, Table 64, Table 65) created and presented in 12.2Appendix C (Critical Loading Cases).for the filtering criteria of 1000, 500, 100, and 50-sized components and for these criteria, 21, 27, 55, and 63 loading cases are selected, respectively.

The validation section will present the effect of the reduced number of 9 and 13 loading cases on the accuracy of the wing box and wing group mass. However, if the accuracy of the reduced number of cases is not sufficient, then more loading case numbers can be considered, as indicated in Table 62, Table 63, Table 64, and Table 65.

In addition to the reduced loading cases analysis, the critical loading cases which drive the sizing of the strut, jury, and offset are presented in Table 33, Table 34, and Table 35

Table 30 Nine critical loading cases, sized at least 5000 wing components (except Case-3) out of 197,367 components in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Load Case Type	Rolling Case Type	Rolling Direction	Manoeuvring In rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Negative	0	100	100	V_B
2	4	52,461.00	Manoeuvring	No Rolling	0.00	0.00	Positive	10,668.00	100.00	100.00	V_C
3	9	6,717.00	Manoeuvring	No Rolling	0.00	0.00	Positive	10,668.00	0.00	0.00	V_C
4	35	7,711.00	Gust	No Rolling	0.00	0.00	Positive	3,048.00	100.00	100.00	V_C
5	50	23,575.00	Gust	No Rolling	0.00	0.00	Positive	3,048.00	0.00	0.00	V_C
6	68	6,760.00	Gust	No Rolling	0.00	0.00	Positive	6,096.00	0.00	0.00	V_C
7	265	14,094.00	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0.00	100.00	100.00	V_C
8	439	25,200.00	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0.00	0.00	0.00	V_C
9	456	16,858.00	Manoeuvring	Steady	Rolling Down	n0	Positive	0.00	100.00	100.00	V_C

Table 31 Thirteen critical loading cases, sized at least 2500 wing components out of 197,367 components in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Negative	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
4	32	3,151	Gust	No Rolling	0	0	Positive	0	0	0	V_C
5	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
6	41	3,618	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C
7	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
8	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
9	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
10	343	3,835	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	0	V_C
11	360	3,939	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_C
12	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
13	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C

Table 32 Reduced number of hybrid loading generated for TBW configurations.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Negative	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
4	18	111	Gust	No Rolling	0	0	Positive	0	100	100	V_D
5	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
6	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
7	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
8	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
9	362	678	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_D
10	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
11	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C
12	458	149	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_D

Table 33 Loading cases which drive the sizing of the struts of the TBW configurations.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	143	Taxi	No Rolling	0	0	Positive	0	100	100	V_B
2	41	19	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C

Table 34 Loading cases which drive the sizing of the offsets of the TBW configurations.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	41	34	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C

Table 35 Loading cases which drive the sizing of the juries of the TBW configurations.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	5	Taxi	No Rolling	0	0	Negative	0	100	100	V_B
2	18	8	Gust	No Rolling	0	0	Positive	0	100	100	V_D
3	362	150	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_D
4	458	8	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_D

The taxi bump cases was expected to steer the sizing of the sturt as presented in the previous studies in the literature (Bradley, Allen and Droney, 2015). It is worth noting that gust load case (Loading Case-41) is also dominant in the sizing of 19 struts of 19 different configurations. This emphasize that in addition to the taxi bump, attention should be given in other loading cases in the sizing of the sturt. The TBW configurations can be analysed with more samples in the future step of this study.

The dominant loading cases which steer the sizing of the juries and the offset are demonstrated in Table 34 and

Table 35. It should be noted that those loading cases are not covered in the reduced number of 9 loading cases; however, they are suggested to be considered in the design studies of TBW configurations. Consequently, a hybrid loading cases is generated for TBW configurations considering the critical loading cases of strut, offset, and juries and these reduced number of loading cases are presented in Table 32. As TBW configurations are limited in the generated aircraft samples compare to the cantilever wings, a further study is suggested for the TBW configurations with more samples.

8.4 Validation of the Reduced Number of Loading Cases

The preceding section detailed six distinct combinations of reduced loading cases. These are tabulated in 12.2Appendix C and are further analysed in this section to evaluate their impact on the precision of wing box and wing group MEs.

Previously discussed in Section - 6.5 (Validation & Comparison), the validation datasets used in this study are presented in Table 55 and Table 56. These datasets, inclusive of actual wing box and wing group mass data from 13 metal wing box cantilever wing designs and one composite wing box TBW, were used for validating the current study.

Figure 97 and Figure 98 present the outcomes of wing box and wing group MEs respectively, comparing them with the actual data via bar charts. For clarity in the labels, the wing box ME method earlier termed as M_WB,NEW-2 is labelled as M_WB,NEW-2,9 LC, Direct in Figure 97, which represents the use of 9 loading cases (as shown in

Table 30) and a direct solver. The label M_WB,NEW-2,9LC,Convergent signifies the use of a convergent solver. Similar logic has been employed to create labels for 13 and 483 loading cases.

In Figure 98, the ME method adopted for the wing group is labelled as M_Wing,NEW-2,Selected. Following the aforementioned labelling logic, labels are assigned for 9, 13, and 483 loading cases, as well as direct or convergent solvers.

To examine the efficacy of the ME methodologies, their error values derived from the validation studies are calculated and presented in Table 36 (wing mass) and Table 37 (wing group mass). As both tables indicate, the use of 9 reduced loading cases for wing box and wing group MEs delivers comparable accuracy to the full loading case method. It is also worth noting that the 9 loading cases comprise the symmetric manoeuvring, combined, gust and, taxi bump.

Table 36 Wing box mass estimation error comparison of 9 and 13 reduced the number of loading cases with the full loading cases analyses using 14 real aircraft data.

	M_WB, NEW-,483 LC,Con	M_WB, NEW-2,483 LC,Dir.	M_WB, NEW-2,9 LC,Con	M_WB, NEW-2,9 LC,Dir.	M_WB, NEW-2,13 LC,Con	M_WB, NEW-2,13 LC,Dir.
SD of the error.	8.26%	8.36%	8.14%	8.23%	8.11%	8.21%
Standard error	2.21%	2.23%	2.18%	2.20%	2.17%	2.19%
Average Error	0.21%	0.20%	-1.25%	-1.31%	-0.78%	-0.84%

Table 37 Wing group mass estimation error comparison of 9 and 13 reduced number of loading cases with the full loading cases analyses using 14 real aircraft data.

	M_WGM, NEW- Selected,483 LC,Con	M_WGM, NEW- Selected,483 LC,Dir.	M_WGM, NEW- Selected,9 LC,Con	M_WGM, NEW- Selected,9 LC,Dir.	M_WGM, NEW- Selected,13 LC,Con	M_WGM, NEW- Selected,13 LC,Dir.
SD of the error.	6.55%	6.49%	6.49%	6.56%	6.48%	6.57%
Standard error	1.75%	1.74%	1.73%	1.75%	1.73%	1.76%
Average Error	-2.13%	-2.22%	-3.12%	-3.15%	-2.83%	-2.87%

Moreover, the application of these 9 reduced loading cases offers a significant computational advantage, reducing computation time by approximately 30 times while maintaining an equivalent level of accuracy. Therefore, these reduced loading cases are selected for subsequent studies.

Finally, it is important to note that for TBW design studies aiming to minimize uncertainties, the hybrid 12 loading cases presented in Table 32 can be utilized.

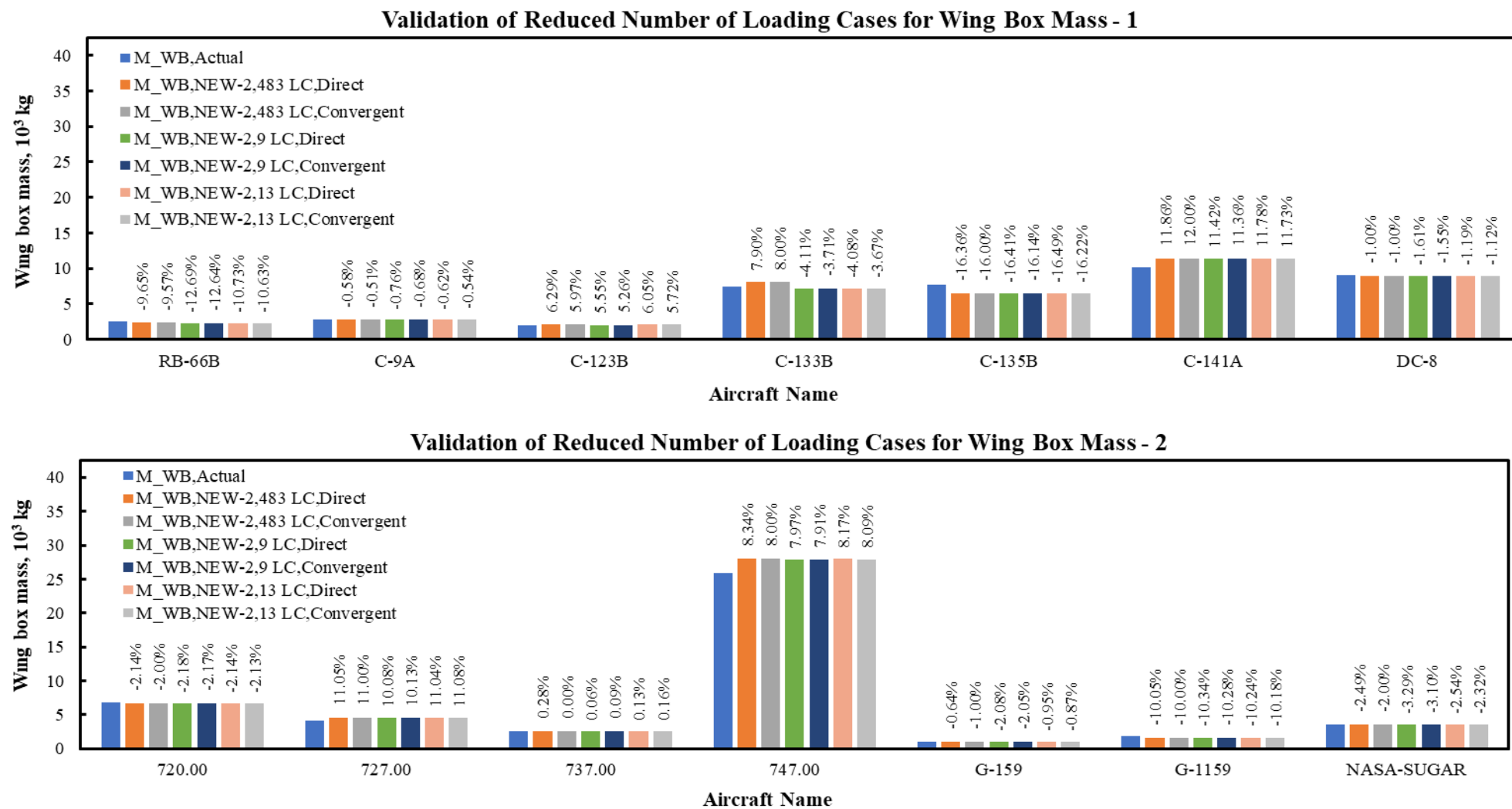
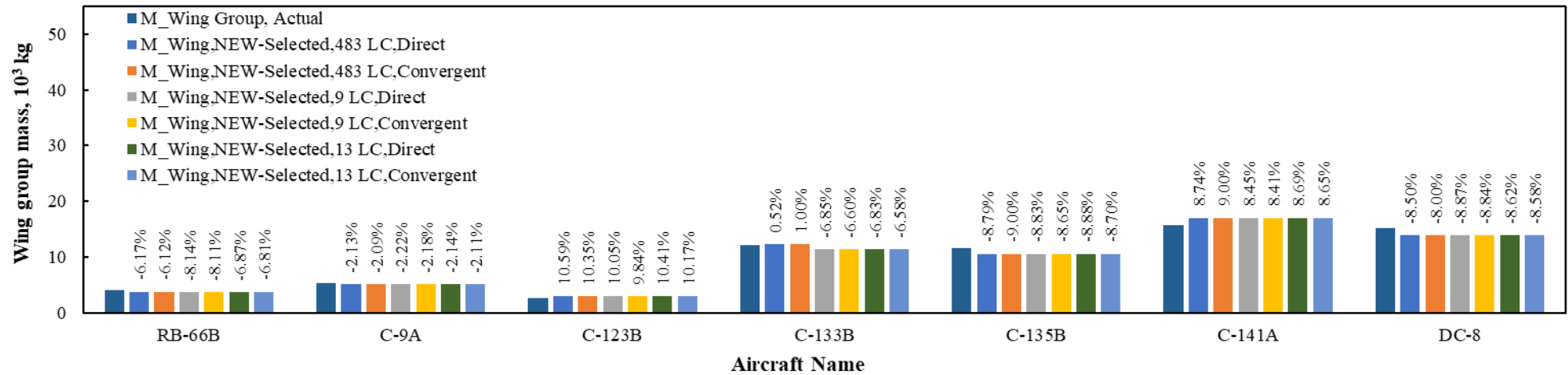


Figure 97 Validation of the reduced number of 9 and 13 loading cases for wing box mass estimation using data from 14 aircraft.

Validation of Reduced Number of Loading Cases for Wing Group Mass - 1



Validation of Reduced Number of Loading Cases for Wing Group Mass - 2

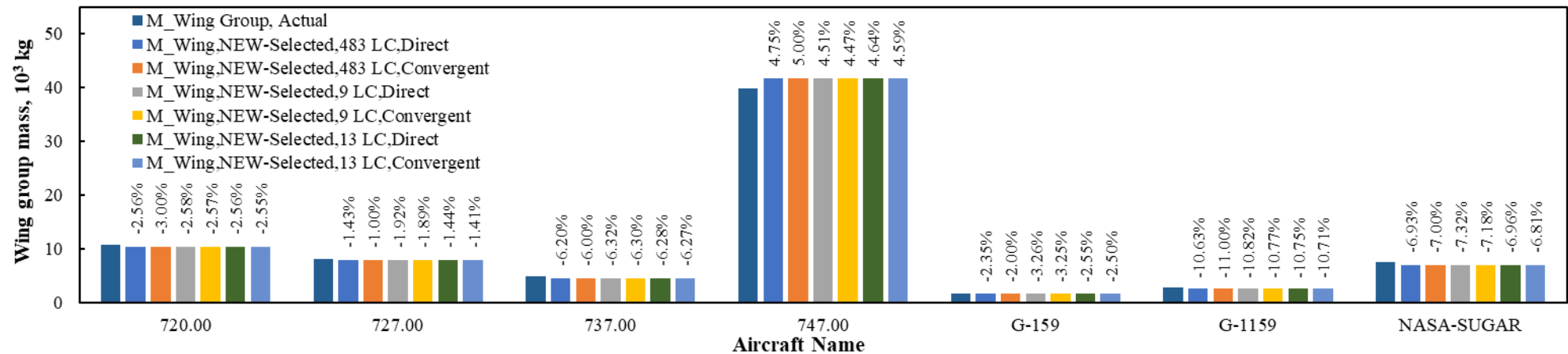


Figure 98 Validation of the reduced number of 9 and 13 loading cases for wing group mass estimation using data from 14 aircraft.

8.4.1 Load Envelopes Comparisons

In the former section, the effectiveness of the reduced number of loading cases was assessed for wing group ME of 14 aircraft. In this section, the analysis is extended by comparing the shear, bending, and torsion envelopes derived from a reduced number of 13 loading cases against those derived from 483 loading cases. The objective is to evaluate the overlap between these envelopes, providing further insight into the reliability of the reduced loading case approach.

For this study, the Boeing 720 aircraft data outlined in Table 56 was employed as the baseline. The wing group mass of this aircraft was estimated using both the exhaustive approach of 483 loading cases and the condensed method of 13 loading cases. Subsequently, the shear load, bending moment, and torsion distributions along the wing were recorded for each approach. Envelopes of these distributions were plotted, capturing their maximum and minimum values, and are presented in Figure 99.

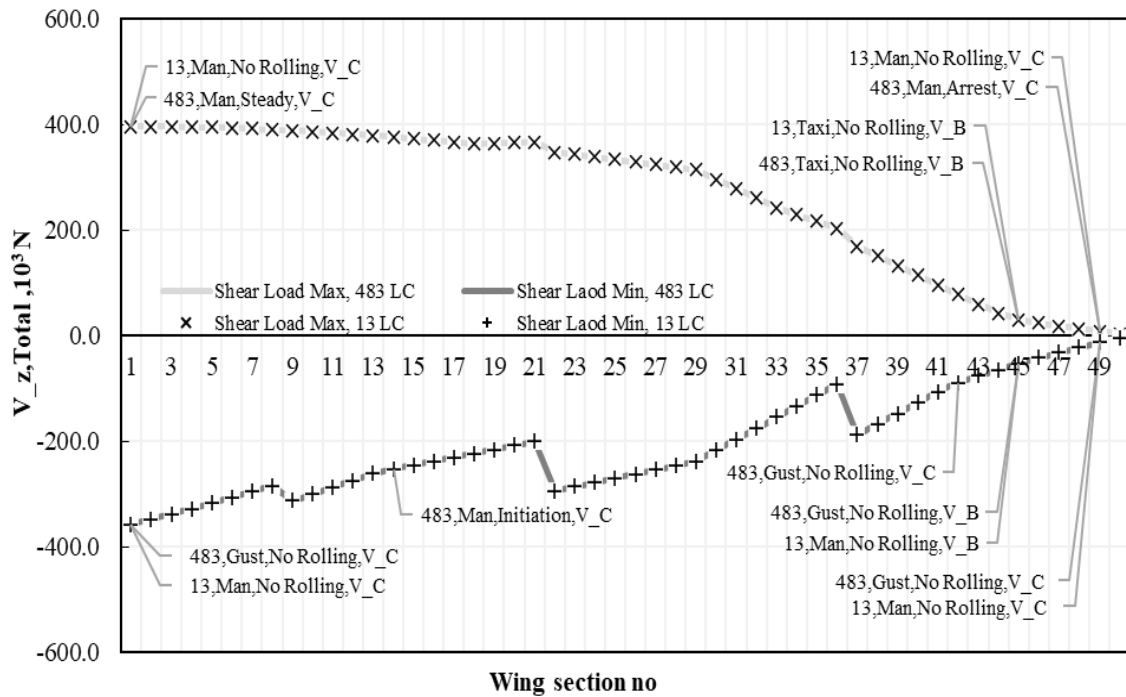
Figure 99 distinguishes between the envelopes derived from the full loading cases (483 cases, represented by continuous lines) and the reduced loading cases (13 cases, denoted by markers). It further provides the details of the dominant loading cases along the wingspan, from root to tip using data labels. To avoid redundancy, changes in dominant cases are highlighted only when they differ from the previous wing section.

Despite the disparity in dominant loading cases for minimum and maximum values of shear load, bending moment, and torsion distribution along the wing when employing 483 and 13 loading cases, a close overlay is observed between the respective envelopes. This suggests that the condensed method, despite its reduced complexity, provides a satisfactory approximation of the full loading case method.

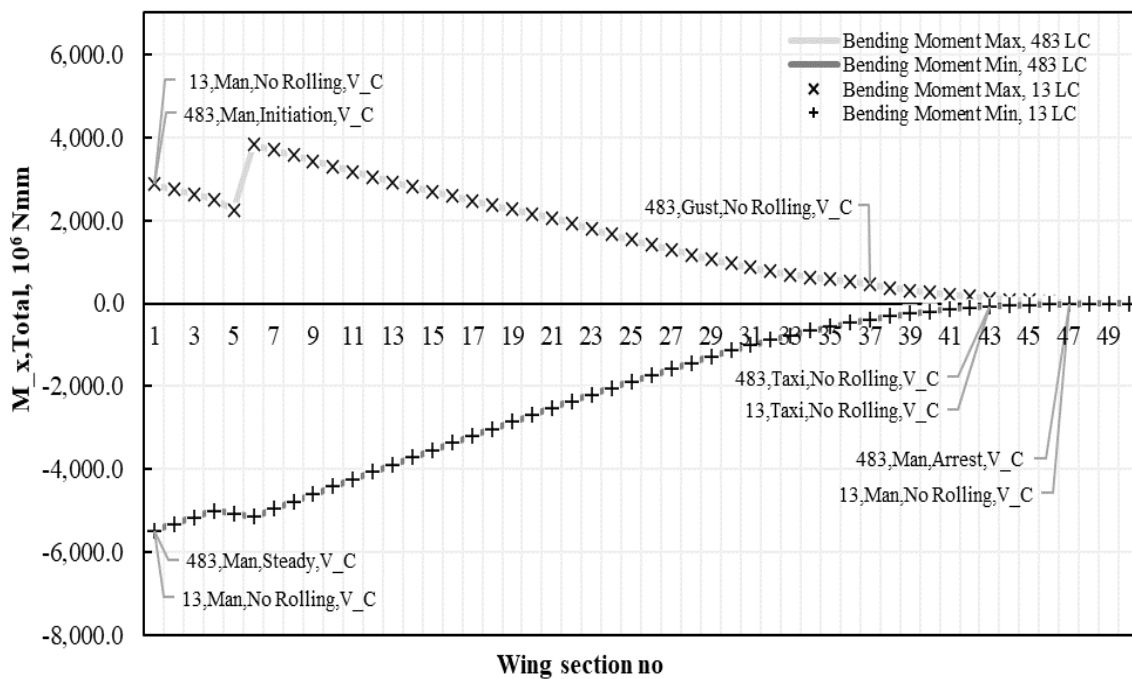
The nuanced comparison of loading cases can be appreciated by referring to Figure 106. The reliable accuracy of the reduced number of loading cases is corroborated by the near congruence of the envelopes in Figure 99, even though the dominant loading case differs for each combination of loading cases.

In summary, this analysis affirms the reliability of the reduced loading case approach in providing an effective yet computationally efficient approximation for shear load, bending moment, and torsion distributions along the wing.

Comparisons of the Shear Load Envelopes



Comparisons of the Bending Moment Envelopes



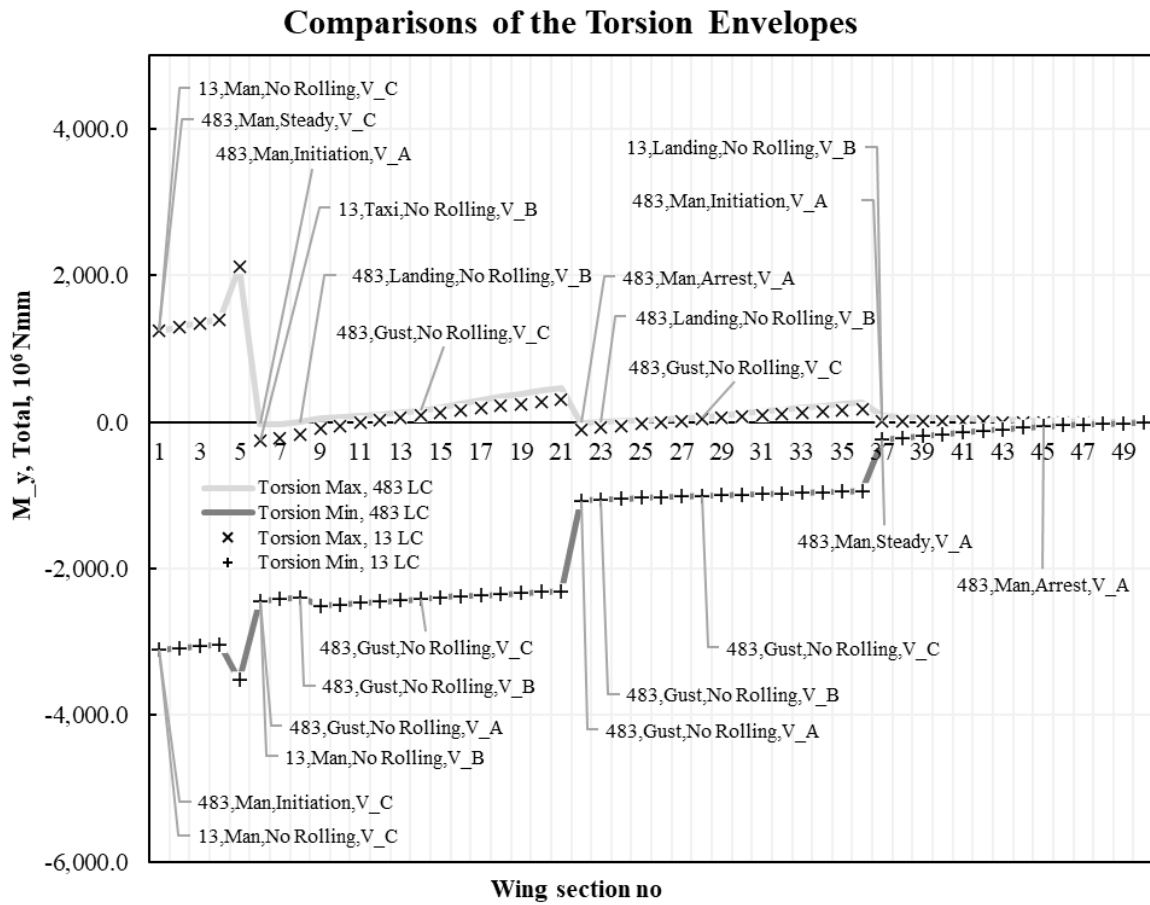


Figure 99 Comparisons of the shear load, bending moment, and the torsion envelopes of 13 and 483 loading cases using Boeing 720.

8.5 Conclusion

This study critically assessed the influence of distinct loading cases on the SSaME of innovative wing designs. These include short AR, moderate AR, and HAR cantilever wings, as well as SBWs and TBWs, integrating either conventional or distributed propulsion, and utilizing hydrogen, electric, or fossil fuels.

To devise a comprehensive list of loading cases accommodating all novel configurations without compromising accuracy in structural sizing, the Latin Hypercube method was employed. This facilitated the generation of 1941 aircraft samples through the manipulation of 71 input variables.

Subsequently, the critical loading cases that guide the sizing of the TBW components, specifically the strut, jury, and offset were identified and outlined.

Different combinations of loading cases, comprising 9, 13, 21, 27, 55, and 63 loading cases, were presented and the methodological steps undertaken to derive the reduced number of loading cases were explained.

Upon analysis, it was found that by employing only 9 critical loading cases, the precision of structural sizing and consequently, the ME of novel wing configurations, was remarkably comparable to that obtained using all loading cases. Thus, these 9 loading cases could effectively replace the full set for preliminary design studies, substantially reducing the computational time by about 30 times compared to the complete set of 483 loading cases.

9 COMPARATIVE and OPTIMISATION STUDIES

The dynamic nature of the aerospace sector has witnessed an array of innovative aircraft wing designs and propulsion systems that significantly impact aircraft weight and efficiency. This chapter undertakes a broad examination of the TBW and cantilever wings with novel propulsion technologies via comparative and optimisation studies.

First, the architectural distinctions among TBW, SBW, and traditional cantilever wing designs were investigated. The intricate variations in their design parameters have a considerable influence on the mass of the wing box and the overall wing.

Next, the effect of propulsion systems on wing mass was investigated with a comparative study. Conventional and distributed propulsion systems were juxtaposed, appraising their impact on various wing designs, including moderate to high AR SBWs, TBWs, and cantilever wings.

Furthermore, use of alternative energy sources, such as hydrogen and electricity, has sparked interest in the location of aircraft fuel storage and the effect of fuel storage location on the wing mass. The differential effects of the traditional wing designs that accommodate fuel in the wing box versus the 'dry wing' design conceived for hydrogen or electric propulsion were assessed.

Lastly, the construction material - a significant factor determining an aircraft wing's weight, strength, and overall performance – was evaluated. The implications of employing quasi-isotropic composite materials against conventional aluminium in the context of the above-mentioned parameters were compared.

After the comparative studies, the optimisation studies were conducted. The objective was to minimise the wing group mass by optimising the design parameters such as three-dimensional locations of concentrated loads and strut-wing attachment.

9.1 Baseline Aircraft

This chapter employs a total of four distinct baseline aircraft—Boeing 737, Boeing 747, TBW (NASA-SUGAR aircraft), and CANT (a cantilever variant of the TBW)—as the foundation for an extensive comparative and optimisation study. These aircraft models are subject to a variety of adaptations concerning the composition of the wing box (metal or composite) and the fuel storage locations for fossil fuel, hydrogen, and electric powered concepts. The specific details of each baseline aircraft, including Boeing 737

and Boeing 747, are contained in Table 56, while the information related to the TBW and CANT aircraft is presented in Table 57.

Each baseline aircraft model is configured in three different ways based on fuel storage. The "Con" designation indicates a conventional approach where fuel is stored in the wing box and fuselage. The "Dry" designation signifies that all fuel is stored within the fuselage. The "1Drop" configuration represents a hybrid approach where half of the fuel normally stored in the wing box is moved to drop tanks, with the remaining fuel stored in the fuselage.

Table 38 The description of the aircraft concepts studied in this chapter.

No	Aircraft Concept	Baseline Name	Wing Box Material	Fuel Storage	Details
1	737, Con,Al	737	Aluminium	Actual Storages/Wing Box and Fuselage	Table 56
2	737,Dry,Al	737	Aluminium	Only in Fuselage	Table 56
3	737,1Drop,Al	737	Aluminium	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 56
4	737,Con,Comp	737	Composite	Actual Storages/Wing Box and Fuselage	Table 56
5	737,Dry,Comp	737	Composite	Only in Fuselage	Table 56
6	737,1Drop,Comp	737	Composite	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 56
7	747,Con,Al	747	Aluminium	Actual Storages/Wing Box and Fuselage	Table 56
8	747,Dry,Al	747	Aluminium	Only in Fuselage	Table 56
9	747,1Drop,Al	747	Aluminium	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 56
10	747,Con,Comp	747	Composite	Actual Storages/Wing Box and Fuselage	Table 56
11	747,Dry,Comp	747	Composite	Only in Fuselage	Table 56
12	747,1Drop,Comp	747	Composite	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 56
13	TBW,Con,Al	TBW	Aluminium	Actual Storages/Wing Box and Fuselage	Table 57
14	TBW,Dry,Al	TBW	Aluminium	Only in Fuselage	Table 57
15	TBW,1Drop,Al	TBW	Aluminium	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 57
16	TBW,Con,Comp	TBW	Composite	Actual Storages/Wing Box and Fuselage	Table 57
17	TBW,Dry,Comp	TBW	Composite	Only in Fuselage	Table 57
18	TBW,1Drop,Comp	TBW	Composite	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 57
19	CANT,Con,Al	CANT	Aluminium	Actual Storages/Wing Box and Fuselage	Table 57
20	CANT,Dry,Al	CANT	Aluminium	Only in Fuselage	Table 57
21	CANT,1Drop,Al	CANT	Aluminium	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 57
22	CANT,Con,Comp	CANT	Composite	Actual Storages/Wing Box and Fuselage	Table 57
23	CAN,Dry,Comp	CANT	Composite	Only in Fuselage	Table 57
24	CAN,1Drop,Comp	CANT	Composite	Half of the wing box fuel moved to Drop Tanks, Rest to Fuselage	Table 57

Moreover, each aircraft is modelled with both aluminium (denoted as "Al") and composite (labelled as "Comp") wing boxes, resulting in six variations per aircraft model, or a total of twenty-four configurations across the four models. These variations, systematically

presented in Table 38, offer a structured overview of the aircraft configurations employed in this study.

9.2 Lift Distribution Models

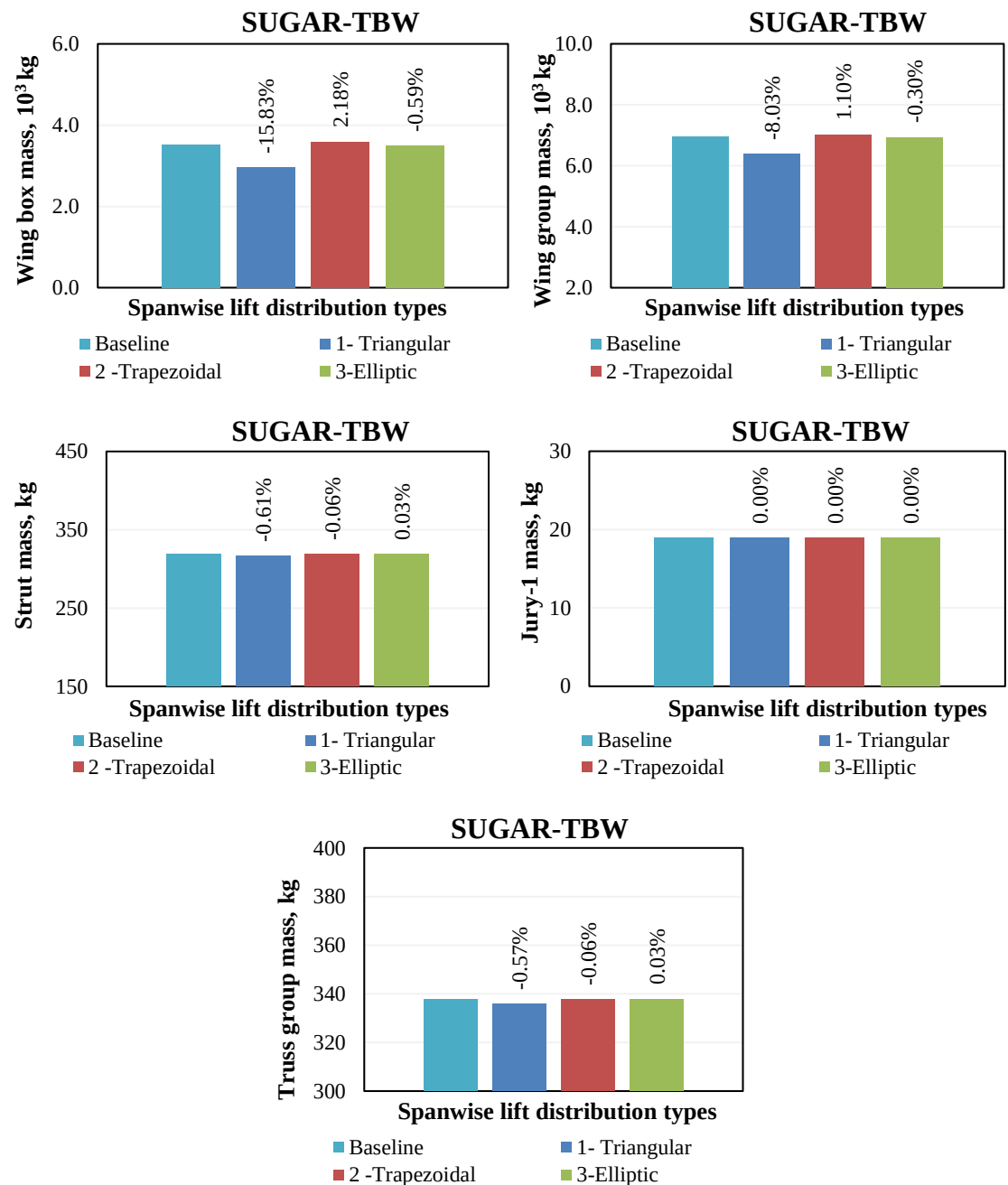


Figure 100 The effect of the lift distribution shape on the mass of the wing group, wing box, strut, and truss of a TBW aircraft.

Simple lift distribution models were explicated in Section - 4.2 (Load Distribution Models) to be applied in the established mathematical model (MM). In this section their effects on

the mass of the structural components of the TBW and CANT aircraft will be elucidated. The baseline concepts have Schrenk's lift distribution over the wing in Figure 100 and Figure 101 for TBW and CANT, respectively.

It can be observed from Figure 100 and Figure 101 that, the lift distribution models primarily influence the wing box mass of the TBW and CANT configurations. Notably, while the Schrenk, elliptic, and trapezoidal distribution models generate nearly congruent ME outcomes, the model utilizing a triangular lift distribution predicts a wing box mass that is 15.83% lighter than that forecasted by the baseline Schrenk model.

Although the studied lift distributions models are inadequate for rigorous aerodynamic performance analysis, they are highly favoured in the conceptual and preliminary structural design studies. The results presented here outline that, the new SSaME method is sensitive to variations in lift distribution shape.

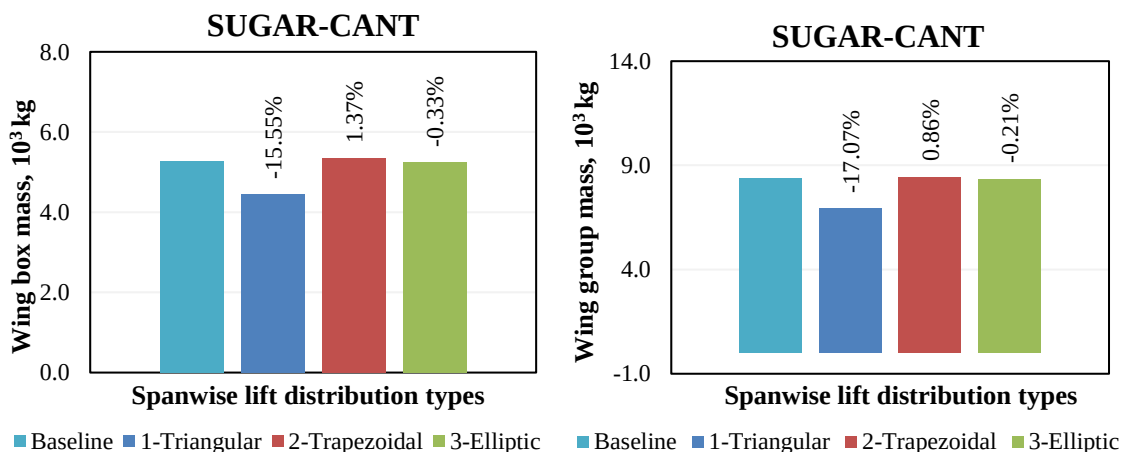


Figure 101 The effect of the lift distribution shape on the mass of the wing group and wing box of the high aspect ratio cantilever wing aircraft.

Moreover, considering the district design of the geometric and structural analyses models presented in this thesis for the wings, the new model offers seamless compatibility with more advanced aerodynamic models for more comprehensive investigations. The future studies can be focused on the optimum wing section design with optimised section sweep, incidence angles, airfoil shape, and other features.

9.3 Cantilever and Truss Braced Wings

In this section, a comparative analysis is conducted to study the impact of the strut on the reduction of wing box and wing group mass. This examination focuses specifically on the aircraft with TBW and cantilever wing.

Table 39 illuminates the MEs of wing boxes for both TBW and CANT aircraft with both metal and quasi-isotropic composite materials. From this analysis, a significant mass reduction becomes evident when considering the transition from the conventional CANT configuration to the TBW design. Specifically, the TBW configurations with both aluminium and composite materials achieved a mass reduction of 27.1% and 33.1% respectively in wing box masses compared to their CANT counterparts.

Table 39 Comparison of the wing box mass estimation of the cantilever and TBWs with metal or quasi-isotropic composite wing boxes.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	CANT,Con,Al	5854.1		8892.4	
2	TBW,Con,Al	4266.4	-27.1%	0.0	
3	CANT,Con,Comp	5266.2		7167.9	
4	TBW,Con,Comp	3523.1	-33.1%	2518.4	-64.9%

In comparison, the Chiozzotto method indicated even more considerable reductions for the same configurations. However, it's important to be cautious when interpreting these figures as the Chiozzotto method tends to overestimate the mass reduction, as is clear from the extreme differences.

Further emphasising the significance of this finding as shown in Table 40 and Table 42, it can also be observed that if a HAR metallic CANT aircraft were to be upgraded with strut support and composite materials, there would be a notable decrease in wing box and wing group masses by 39.8% and 21.6% respectively.

Table 40 Wing box mass reduction of the TBW with a composite wing box compared to the cantilever wing with a metal wing box.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	CANT,Con,Al	5854.1		8892.4	
2	TBW,Con,Comp	3523.1	-39.8%	2518.4	-71.7%

Table 41 Comparison of the wing group mass estimation of the cantilever and TBWs with metallic or quasi-isotropic composite wing boxes.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	CANT,Con,Al	8970.2		11906.5	
2	TBW,Con,Al	7686.4	-14.3%		
3	CANT,Con,Comp	8382.3		10182.1	
4	TBW,Con,Comp	7035.0	-16.1%	7083.6	-30.4%

Table 42 Wing group mass reduction of the TBW with a composite wing box compared to the cantilever wing with a metallic wing box.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	CANT,Con,Al	8970.2		11906.5	
2	TBW,Con,Comp	7035.0	-21.6%	7083.6	-40.5%

Table 43 then provides a breakdown of the mass of the wing group for both TBW and CANT configurations. This table reveals a significant reduction in the bending resistant material mass in the TBW configuration, a benefit attributable to the addition of the strut. Although the strut introduces a minimal additional mass, the overall wing group mass for the TBW configuration is still less than that of the CANT configuration. This suggests that the benefits offered by the strut support outweigh its associated mass cost, and indeed, the inclusion of a strut proves to be a beneficial alteration to aircraft design.

Table 43 The effect of struts on the wing group masses with composite and aluminium wing boxes.

Aircraft	Shear Resistant Material, kg	Bending Resistant Material, kg	Rib, kg	Wing Box Other, kg	Wing Other, kg	Jury, kg	Strut, kg	Total Wing Group Mass, kg	Mass Save, %
CANT,Con,Al	1,714.0	2,869.1	221.5	1,049.4	3,115.9		0.0	8,970.0	
TBW,Con,Al	1,598.0	1,397.4	221.5	1,049.4	3,115.9	13.4	300.4	7,696.0	-14.2%
CANT,Con,Comp	1,087.3	3,235.8	125.0	818.0	3,115.9		0.0	8,382.0	
TBW,Con,Comp	1,010.3	1,569.8	125.0	818.0	3,115.9	13.7	392.3	7,044.9	-16.0%

In conclusion, the analysis emphasizes the potential benefits of incorporating a strut into aircraft wing design, particularly for those using a TBW configuration. This addition leads to a significant mass reduction in both wing box and wing group components and enhances structural efficiency in handling shear and bending forces. These findings suggest a need for further investigation and consideration of strut-supported wing designs in future aircraft configurations.

9.4 Conventional and Distributed Propulsion

This section investigates the effect of distributed propulsion in the wing box and wing group mass. For this study baseline aircraft of Boeing 747, Boeing 737, TBW and CANT are used. Their details are given in Section - 9.1 (Baseline Aircraft). Except Boeing 747, other aircraft have two wing mounted engines while Boeing 747 have four engines. The MM developed in this thesis, requires the total mass and thrust of the engines mounted in half wing to compute the loads they cause on the structures.

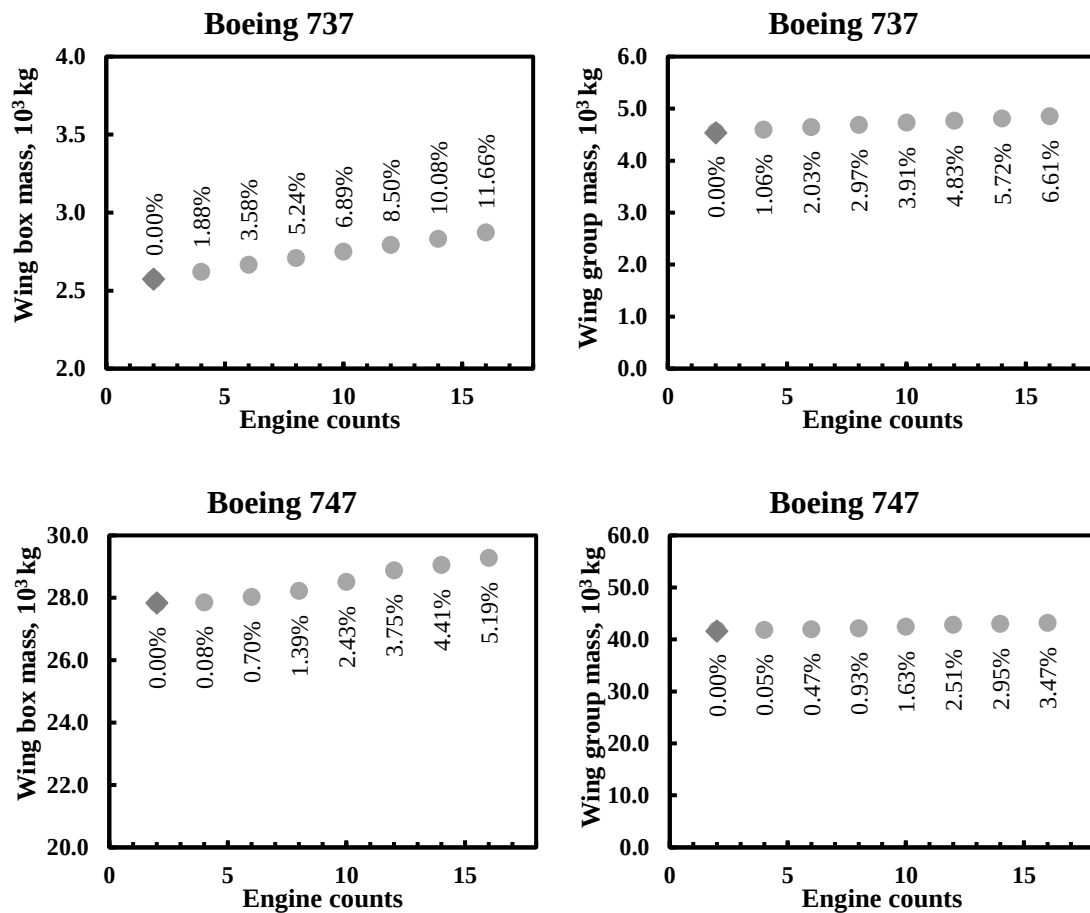


Figure 102 The effects of the distributed engines on the wing box and wing group mass of Boeing 737 and 747 aircraft.

A simple approach is utilised to compute the mass and thrust of the distributed engines. It was assumed that if the total mass and thrust of the original engines are $M_{Engine,2}, M_{Engine,2} \dots T_{Engine,2}, T_{Engine,2}$, respectively, the mass and thrust of each engine in the distributed engine configuration of these baseline aircraft can be computed with $(M_{Engine,2}, M_{Engine,2} \dots)/N$ and $(T_{Engine,2}, T_{Engine,2} \dots)/N$ where N is the total number of engines in the distributed configuration. The engines' position in the z and x axes remains

the same and equal to the original configuration, while their spanwise position is found using a simple assumption. It was assumed that the engines are located in a spanwise position with an equal pitch. Therefore, the position of each engine can be computed with $(b/2)/(\frac{N}{2} + 1)$ on a half wing. Hence in a two-engine configuration, for instance, the engines are located in the middle of the wing. The mass and thrust of the original engines of the configurations are resized from 2 engine configurations to 16 engines.

Boeing 737 and 747 are moderate AR aircraft compared to HAR aircraft of TBW and CANT. It is evident from Figure 102 that, the wing box and group mass of moderate AR Boeing 737 and 747 are increased with increasing engine counts. The minimum engine number for both configurations were two engines.

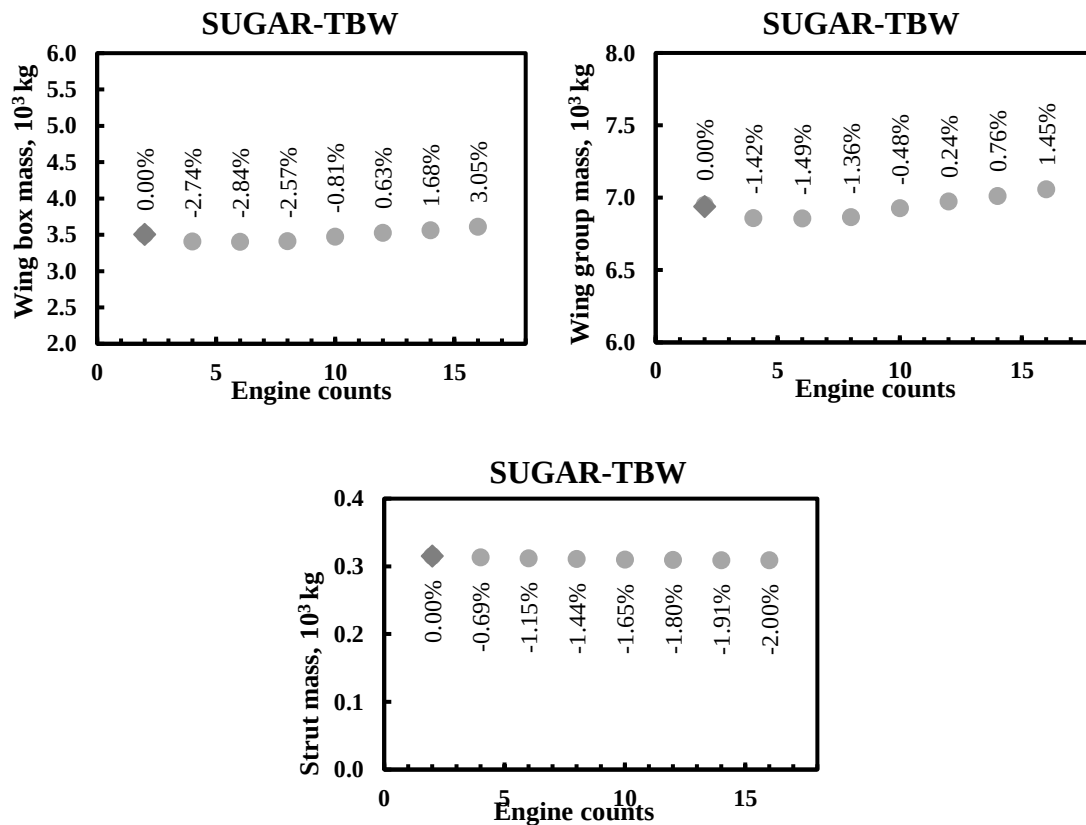


Figure 103 The effects of the distributed engines on the wing box, strut and wing group mass of TBW aircraft.

Conversely, HAR CANT and TBW aircraft reached minimums of wing box and wing group mass with a 6 engines configuration as depicted in Figure 104. The mass reductions of the wing group mass of CANT and TBW are 1.49% and 1.12% with a six engines configuration compared to the two engines one.

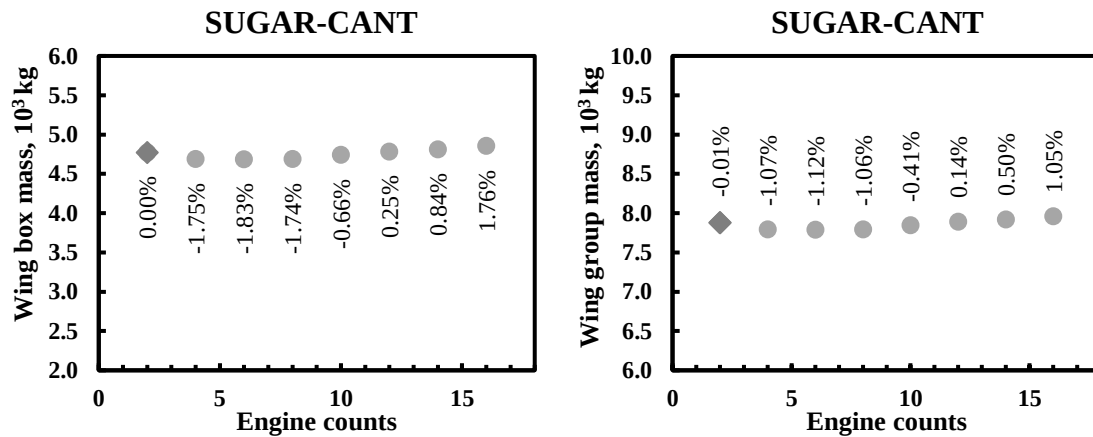


Figure 104 The effects of the distributed engines on the wing box and wing group mass of CANT aircraft.

This study indicates that the wing box and wing group mass behaviour of moderate AR and HAR aircraft are different to increasing engine counts. While the increasing engine counts raise the wing box and wing group mass of moderate AR wings notably, they cause a minimal decline in HAR wings with or without a strut. Further studies are required covering the single and multidisciplinary optimisation approaches.

9.5 Conventional and Dry Wings

Conventionally, wing boxes, fuselages, and drop tanks have been the primary repositories for fossil fuels in conventional aircraft designs. With the increasing popularity of the electric and hydrogen powered aircraft, new design concepts have been under consideration. Markedly, hydrogen-fuelled aircraft are inclined towards fuselage-based fuel storage due to the inherent complexities and limitations associated with storing hydrogen. Therefore, the wing box spaces without fuel tanks lead to the emergence of “dry wings”.

The ME method studied in this thesis was developed considering these technologies and it can be employed in the SSaME of the dry wings as it relies on the analytical structural sizing methods.

Formerly, the effect of the mass of the fuel stored in the wing on the mass of the wing box of baseline aircraft of CANT, TBW, 737 and 747 were studied in Section - 7.2.4.2 (Fuel Stored in the Wing). Following this study the spar positions of a dry wing was also investigated in Section - 7.2.3.2 (ND Distance Between the Spar) as they have more space to be located in the absence of the internal wing fuel tanks. This section extends

the work on dry wing by considering the different wing box materials and wing drop tanks. Hence, the effect of dry wing design on the wing group and box masses of 24 aircraft configuration, whose details are described in Table 38, will be investigated.

Table 44 Comparisons of the wing group mass estimations of the conventional wing, dry wing, and dry wing with a drop tank.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	737, Con,Al	4548.6		5120.3	
2	737,Dry,Al	4704.1	3.4%	5120.3	0.0%
3	737,1Drop,Al	4576.4	0.6%	5120.3	0.0%
4	737,Con,Comp	3826.4		4675.0	
5	737,Dry,Comp	4126.8	7.9%	4675.0	0.0%
6	737,1Drop,Comp	3902.3	2.0%	4675.0	0.0%
7	747,Con,Al	41767.8		47328.8	
8	747,Dry,Al	45838.0	9.7%	47328.8	0.0%
9	747,1Drop,Al	42812.3	2.5%	47328.8	0.0%
10	747,Con,Comp	35917.2		43957.2	
11	747,Dry,Comp	42575.3	18.5%	43957.2	0.0%
12	747,1Drop,Comp	37373.7	4.1%	43957.2	0.0%
13	TBW,Con,Al	7686.4			
14	TBW,Dry,Al	7886.9	2.6%		
15	TBW,1Drop,Al	7795.0	1.4%		
16	TBW,Con,Comp	7035.0		7083.6	
17	TBW,Dry,Comp	7245.9	3.0%	7083.6	0.0%
18	TBW,1Drop,Comp	7169.2	1.9%	7083.6	0.0%
19	CANT,Con,Al	8970.2		11906.5	
20	CANT,Dry,Al	9697.9	8.1%	11906.5	0.0%
21	CANT,1Drop,Al	9173.3	2.3%	11906.5	0.0%
22	CANT,Con,Comp	8382.3		10182.1	
23	CAN,Dry,Comp	9184.7	9.6%	10182.1	0.0%
24	CAN,1Drop,Comp	8626.5	2.9%	10182.1	0.0%

Primarily, it is evident from Table 44 and Table 45 that the novel estimation method devised in this study exhibits higher sensitivity and accuracy compared to existing methodologies, as it conforms to the logical expectation that dry wings, without fuel mass relief provided by fuel stored within the wing boxes, should exhibit greater mass. Conversely, existing methodologies presented either contrary results or exhibited diminished sensitivity to this factor.

Comparative analysis of conventional, dry and drop tank supplemented wing configurations across various aircraft revealed a consistent trend. Dry wing configurations, regardless of the aircraft type or wing material (Aluminium or Composite), demonstrated an increased ME percentage when compared to their conventional

counterparts. Furthermore, the addition of a drop tank led to a marginal reduction in this mass increase, suggesting that the presence of a drop tank could partially compensate for the absence of in-wing fuel storage in dry wings.

Table 45 Comparison of the wing box mass estimation of the conventional wing, dry wing and dry wing with a drop tank.

No	Aircraft	NEW Method	Differ. %	P. York	Differ. %	Chiozzotto	Differ. %
1	737, Con,Al	2,588.8		2,815.1		3,094.8	
2	737,Dry,Al	2,744.3	6.0%	2,697.9	-4.2%	3,094.8	0.0%
3	737,1Drop,Al	2,616.6	1.1%	2,700.1	-4.1%	3,094.8	0.0%
4	737,Con,Comp	1,866.7		2,815.1		2,649.5	
5	737,Dry,Comp	2,167.1	16.1%	2,697.9	-4.2%	2,649.5	0.0%
6	737,1Drop,Comp	1,942.5	4.1%	2,700.1	-4.1%	2,649.5	0.0%
7	747,Con,Al	27,966.7		27,454.0		33,021.8	
8	747,Dry,Al	32,036.9	14.6%	27,048.3	-1.5%	33,021.8	0.0%
9	747,1Drop,Al	29,011.2	3.7%	27,067.5	-1.4%	33,021.8	0.0%
10	747,Con,Comp	22,116.1		27,454.0		29,650.2	
11	747,Dry,Comp	28,774.2	30.1%	27,048.3	-1.5%	29,650.2	0.0%
12	747,1Drop,Comp	23,572.6	6.6%	27,067.5	-1.4%	29,650.2	0.0%
13	TBW,Con,Al	4,266.4		6,593.1		0.0	
14	TBW,Dry,Al	4,394.4	3.0%	6,467.6	-1.9%	0.0	
15	TBW,1Drop,Al	4,362.4	2.3%			0.0	
16	TBW,Con,Comp	3,523.1		6,592.9		2,518.4	
17	TBW,Dry,Comp	3,625.5	2.9%	6,467.6	-1.9%	2,518.4	0.0%
18	TBW,1Drop,Comp	3,638.6	3.3%	6,472.1	-1.8%	2,518.4	0.0%
19	CANT,Con,Al	5,854.1		6,592.9		8,892.4	
20	CANT,Dry,Al	6,581.8	12.4%	6,467.6	-1.9%	8,892.4	0.0%
21	CANT,1Drop,Al	6,057.2	3.5%	6,472.1	-1.8%	8,892.4	0.0%
22	CANT,Con,Comp	5,266.2		6,593.1		7,167.9	
23	CAN,Dry,Comp	6,068.6	15.2%	6,467.8	-1.9%	7,167.9	0.0%
24	CAN,1Drop,Comp	5,510.4	4.6%	6,472.2	-1.8%	7,167.9	0.0%

In terms of wing material, quasi-isotropic composite wings consistently showed higher percentage increases in wing group mass estimates for dry wings, as compared to their metallic counterparts, across all aircraft types. This could be attributed to the inherently lighter nature of composite wings, making them more sensitive to the lack of in-wing fuel storage mass relief.

High aspect ratio aircraft such as TBW and CANT showed somewhat variable responses to the dry wing configuration. For TBW aircraft, the wing group and box mass raise for both aluminium and composite wing materials were relatively moderate. Inversely, CANT aircraft demonstrated higher percentage increases.

Although there is not a clear correlation between the wing group and box masses of different configurations, the highest increase in the mass of those components are seen in Boeing 747.

The TBW configuration, which incorporates struts and jury, demonstrates a minimal percentage increase in mass estimates for the dry wing configuration compared to the other configurations, thus indicating the structural advantage of this concept for dry wing designs.

In conclusion, the presented results underline the profound influence of wing design (conventional vs dry), wing material (Aluminium vs Composite), and aircraft configuration on the mass estimates of the wing group and wing box. The findings reiterate the sensitivity and accuracy of the new ME method, thereby emphasizing its potential for application in future aircraft design endeavours. Future research can further delve into exploring the optimisation of various aircraft configurations with different fuel storage options considering the results presented in this section and in the former sections of 7.2.4.2 and 7.2.3.2.

9.6 Metallic and Composite Wing Boxes

This section examines the influence of the metallic and quasi-isotropic composite materials on the wing group and wing box mass of the 24 aircraft design explained in Section - 9.1 (Baseline Aircraft). Table 46 and Table 47 compare the mass-saving percentages in the wing group and box masses to observe if any configuration is more sensitive to the wing structural material.

The findings suggest a significant potential for mass reduction when using quasi-isotropic composite materials, as seen in Table 46 and Table 47 for wing group and wing box masses, respectively. This transition to lighter materials has implications not only for the performance and fuel efficiency of the aircraft but also for cost-effectiveness and environmental impact.

The application of the Chiozzotto method led slightly different mass estimation compared to the new method. This discrepancy might be due to the different assumptions and parameters used in the two methods.

Table 46 Comparison of the effect of the quasi-isotropic composite and metal alloys on the wing group mass.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	737, Con,Al	4548.6		5120.3	
2	737,Con,Comp	3826.4	-15.9%	4675.0	-8.7%
3	737,Dry,Al	4704.1		5120.3	
4	737,Dry,Comp	4126.8	-12.3%	4675.0	-8.7%
5	737,1Drop,Al	4576.4		5120.3	
6	737,1Drop,Comp	3902.3	-14.7%	4675.0	-8.7%
7	747,Con,Al	41767.8		47328.8	
8	747,Con,Comp	35917.2	-14.0%	43957.2	-7.1%
9	747,Dry,Al	45838.0		47328.8	
10	747,Dry,Comp	42575.3	-7.1%	43957.2	-7.1%
11	747,1Drop,Al	42812.3		47328.8	
12	747,1Drop,Comp	37373.7	-12.7%	43957.2	-7.1%
13	TBW,Con,Al	7686.4			
14	TBW,Con,Comp	7035.0	-8.5%	7083.6	
15	TBW,Dry,Al	7886.9			
16	TBW,Dry,Comp	7245.9	-8.1%	7083.6	
17	TBW,1Drop,Al	7795.0			
18	TBW,1Drop,Comp	7169.2	-8.0%	7083.6	
19	CANT,Con,Al	8970.2		11906.5	
20	CANT,Con,Comp	8382.3	-6.6%	10182.1	-14.5%
21	CANT,Dry,Al	9697.9		11906.5	
22	CAN,Dry,Comp	9184.7	-5.3%	10182.1	-14.5%
23	CANT,1Drop,Al	9173.3		11906.5	
24	CAN,1Drop,Comp	8626.5	-6.0%	10182.1	-14.5%

Table 47 Comparison of the effect of the quasi-isotropic composite and metal alloy materials on the wing box mass.

No	Aircraft	NEW Method	Differ. %	Chiozzotto	Differ. %
1	737, Con,Al	2588.8		3094.8	
2	737,Con,Comp	1866.7	-27.9%	2649.5	-14.4%
3	737,Dry,Al	2744.3		3094.8	
4	737,Dry,Comp	2167.1	-21.0%	2649.5	-14.4%
5	737,1Drop,Al	2616.6		3094.8	
6	737,1Drop,Comp	1942.5	-25.8%	2649.5	-14.4%
7	747,Con,Al	27966.7		33021.8	
8	747,Con,Comp	22116.1	-20.9%	29650.2	-10.2%
9	747,Dry,Al	32036.9		33021.8	
10	747,Dry,Comp	28774.2	-10.2%	29650.2	-10.2%
11	747,1Drop,Al	29011.2		33021.8	
12	747,1Drop,Comp	23572.6	-18.7%	29650.2	-10.2%
13	TBW,Con,Al	4266.4		0.0	
14	TBW,Con,Comp	3523.1	-17.4%	2518.4	
15	TBW,Dry,Al	4394.4		0.0	
16	TBW,Dry,Comp	3625.5	-17.5%	2518.4	
17	TBW,1Drop,Al	4362.4		0.0	
18	TBW,1Drop,Comp	3638.6	-16.6%	2518.4	
19	CANT,Con,Al	5854.1		8892.4	
20	CANT,Con,Comp	5266.2	-10.0%	7167.9	-19.4%
21	CANT,Dry,Al	6581.8		8892.4	
22	CAN,Dry,Comp	6068.6	-7.8%	7167.9	-19.4%
23	CANT,1Drop,Al	6057.2		8892.4	
24	CAN,1Drop,Comp	5510.4	-9.0%	7167.9	-19.4%

9.7 Single Disciplinary Optimisation Studies for Minimum Wing Group Mass

In the previous sections and chapters of the thesis, the satisfactory sensitivity and accuracy of the new structural design and mass estimation (SDaME) method are tested with several validation, verification, and parametric studies. In addition to its accuracy and sensitivity level for preliminary design studies, the architecture of the MM makes it suitable to be used in single and multi-disciplinary design optimisation works. This shows the optimisation capabilities of the developed model with sample single disciplinary applications. The single disciplinary optimisation means that only the constraints presented in this thesis for the MM are covered. Minimum wing group mass is selected

as an optimisation objective and NASA-SUGAR aircraft is utilised as a baseline TBW aircraft (Bradley, Allen and Droney, 2015).

Most of the aircraft design parameters of the new model can be optimised using available optimisation methods in the literature. An optimisation frame was developed using global and local optimisation algorithms available in python libraries. In the following sections, the details of the algorithms and optimisation works are explained.

The author would like to warn the reader that the primary objective of the optimization study is the minimum wing group mass. And this is not a multidisciplinary optimization of the complete aircraft. This section is crucial for demonstrating the application of the developed method in such optimization studies, even if they are single disciplinary. The results not only illustrate the trend for achieving the minimum wing group mass objective by varying the engine location but also explore the impact of engine count along with other variables. The section investigates this aspect instead of optimizing the entire aircraft. The author believes that these findings are valuable; however, it is advisable to validate these outcomes by comparing them with higher fidelity results before their incorporation, as the current results aim to provide insights primarily into conceptual design rather than detailed design.

9.7.1 Optimisation Methods

In this work, Differential Evolution (DE), a global optimizer, and Sequential Least Squares Programming (SLSQP), a local optimizer, were both used as optimisation techniques from the Python library.

Given that the wing design optimisation problem is expected to provide several local optima, differential evolution optimisation was largely used to ensure a thorough exploration of the solution space. DE is a stochastic, population-based algorithm that enhances the candidate solutions iteratively through operations similar to genetic crossover, mutation, and selection. In the literature, DE's advantage in resolving challenging multidimensional optimisation issues is well established (Storn and Price Kenneth, 1997).

The SLSQP local optimiser can be implemented once the global optimisation is finished. SLSQP is a member of the family of gradient-based techniques that uses the objective function's derivative to direct the search for the best solution. This technique performs exceptionally well in problems with continuous, differentiable, linear, or nonlinear

constraints and is particularly well suited to medium-sized optimisation problems (Kraft, 1988).

9.7.2 Optimisation of the Strut Attachment Location

In this step, the spanwise and the chordwise positions of the strut-wing attachment point was optimised using DE global optimiser only. The lower, upper bounds and, the optimised values of the variables are given in Table 48. The upper and lower bounds in chordwise direction are chosen considering the ND chordwise positions of the front and the rear spars.

The strut position was 58% span length in the baseline design obtained from a detailed multidisciplinary study with multidisciplinary constraints (Bradley, Allen and Droney, 2015). The optimiser using the MM developed in this thesis located the strut attachment at 54% span length for minimum wing group mass objective. The result is very close to the reference study. This outcome also illustrates the robustness of the developed MM for a wide range of design exploration and optimisation studies.

Table 48 Two optimised variables with the upper and lower bounds.

No	Parameters	Lower Bound	Optimised Values	Upper Bound
1	Λ_{strut}	0.20	0.54	0.99
2	$\epsilon_{\text{strut},i}$	0.15	0.25	0.65

Table 49 presents the estimated mass values of strut, jury, wing box and wing of the baseline and optimised (as given in Figure 105) TBWs using the new MM. The optimised mass values of the baseline TBW is very close to the baseline TBW. These results verify the accuracy of the optimiser and the suitability of the new MM in the optimisation studies.

Table 49 Comparison of the estimated mass values of the baseline TBW and optimised TBW with two variables.

	Strut, kg	Jury, kg	Wing Box, kg	Wing, kg
Baseline Estimated Results	486.63	13.72	3,523.13	7,129.41
Baseline Optimised Results	401.57	11.76	3,525.95	7,045.21
Difference, %	-17.48%	-14.29%	0.08%	-1.18%

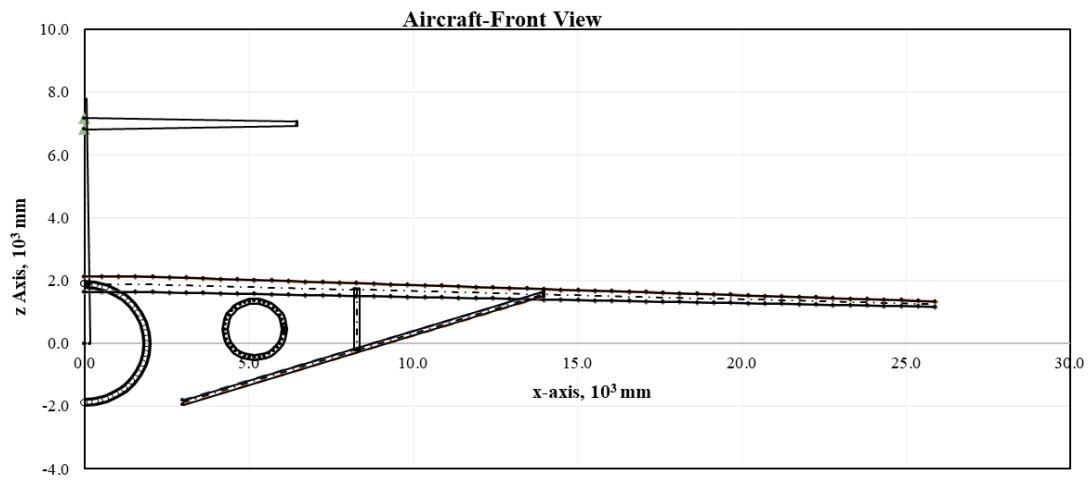
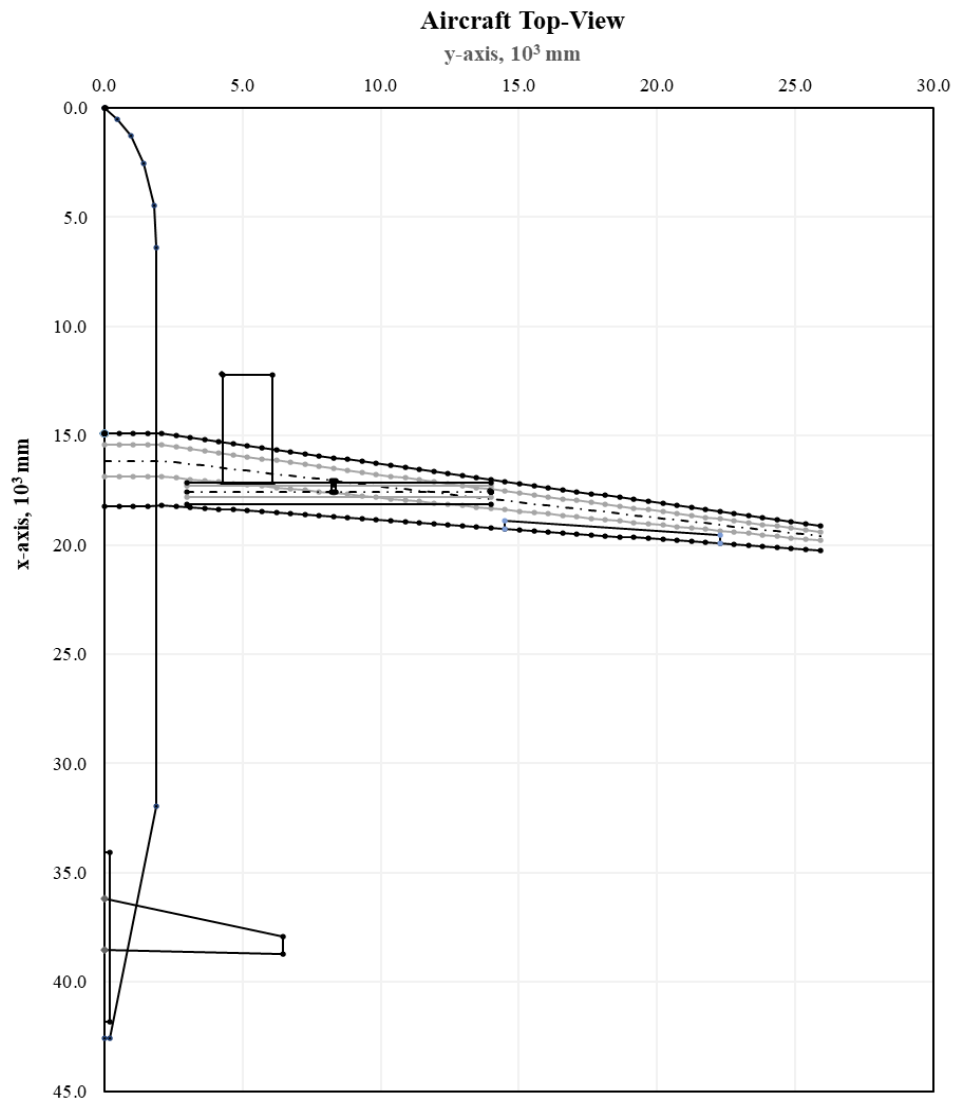


Figure 105 2D top and the front views of the optimised TBW with two optimisation variables.

9.7.3 Location Optimisation of the Strut Attachment and Two Engines

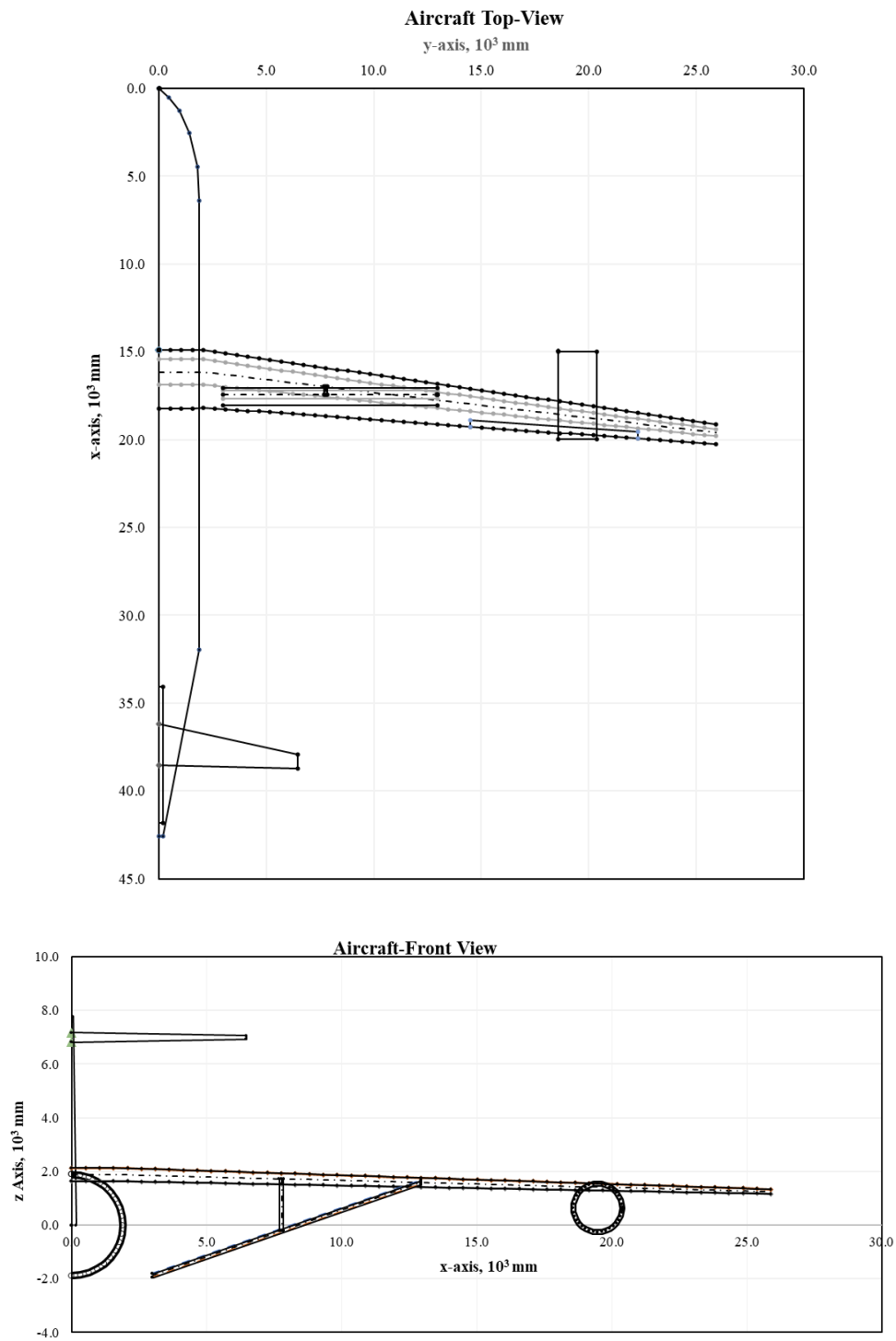


Figure 106 2D top and the front views of the optimised TBW with three optimisation variables.

A similar optimisation study to the one in the previous section is conducted with three variables in this section. In addition to the strut attachment points, the spanwise position of the engine is also optimised for the minimum wing group mass. The optimum single engine position on a TBW was investigated for minimum wing group mass objective. The upper, lower bounds, and the optimised values are presented in Table 50.

Table 50 Three optimised variables with the upper and lower bounds.

No	Parameters	Lower Bound	Optimised Values	Upper Bound
1	$\Lambda_{\text{Engine},1}$	0.15	0.75	0.99
2	Λ_{strut}	0.20	0.50	0.99
3	$\varepsilon_{\text{strut},i}$	0.15	0.27	0.65

The optimised locations of the engine and the strut can be seen from Table 50 and Figure 106. The spanwise strut attachment and engine are located at 50% and 75% wingspan length, respectively. The optimised configuration increased the wing group mass savings even further compared to the optimised configuration given in the previous section in Table 48. The optimised configuration with three variables increased the wing group mass saving from -1.18% to -7.18% as can be seen from Table 49 and Table 51, respectively.

Table 51 Comparison of the estimated mass values of the baseline TBW and optimised TBW with three variables.

	Strut, kg	Jury, kg	Wing Box, kg	Wing, kg
Baseline Estimated Results	486.63	13.72	3,523.13	7,129.41
Baseline Optimised Results	146.99	12.33	3,352.01	6,617.26
Difference, %	-69.79%	-10.15%	-4.86%	-7.18%

This findings also match with the previous studies (Grasmeyer, 1999). Grasmeyer (1999) showed that the wing mass of SBW configuration decreases when the engine located towards to the wing tip. However this configuration also increased vertical tail area in a single engine-off flight case and it was not optimum design for fuel consumption (Grasmeyer, 1999).

9.7.4 Location Optimisation of the Strut Attachment and Sixteen Engines

This study investigates the optimisation of a TBW wing with 16 engines. The engine mass and the trust of the single engine of the baseline was distributed equally in 8 engines for the half wing.

The optimised locations of each engine and the strut attachment are given in Table 52 and Figure 107. The strut spanwise position moved more inboard location of 38% wingspan length. And the distributed engines are located between 60% and 92% wingspan length. The minimum wing group mass for this optimised configuration was 6,810.84kg which is -4.47% lower compared to the baseline configuration as depicted in Table 53.

Table 52 Ten optimised variables with the upper and lower bounds.

No	Parameters	Lower Bound	Optimised Values	Upper Bound
1	$\Lambda_{_Engine,1}$	0.15	0.88	0.99
2	$\Lambda_{_Engine,2}$	0.15	0.64	0.99
3	$\Lambda_{_Engine,3}$	0.15	0.82	0.99
4	$\Lambda_{_Engine,4}$	0.15	0.92	0.99
5	$\Lambda_{_Engine,5}$	0.15	0.84	0.99
6	$\Lambda_{_Engine,6}$	0.15	0.69	0.99
7	$\Lambda_{_Engine,7}$	0.15	0.60	0.99
8	$\Lambda_{_Engine,8}$	0.15	0.80	0.99
9	$\Lambda_{_strut}$	0.20	0.38	0.99
10	$\varepsilon_{_strut,i}$	0.15	0.33	0.65

Table 53 Comparison of the estimated mass values of the baseline TBW and optimised TBW with ten variables.

	Strut, kg	Jury, kg	Wing Box, kg	Wing, kg
Baseline Estimated Results	486.63	13.72	3,523.13	7,129.41
Baseline Optimised Results	125.38	11.90	3,567.63	6,810.84
Difference, %	-13.29%	-74.24%	1.26%	-4.47%

This outcome indicates that the 16-engine configuration is not the optimum for minimum wing group mass; however, this configuration might be achieved with minimal mass penalties compared to the TBW with two engine configurations. This primary finding indicates that future studies are required to investigate the TBW with distributed propulsion in a MDO environment.

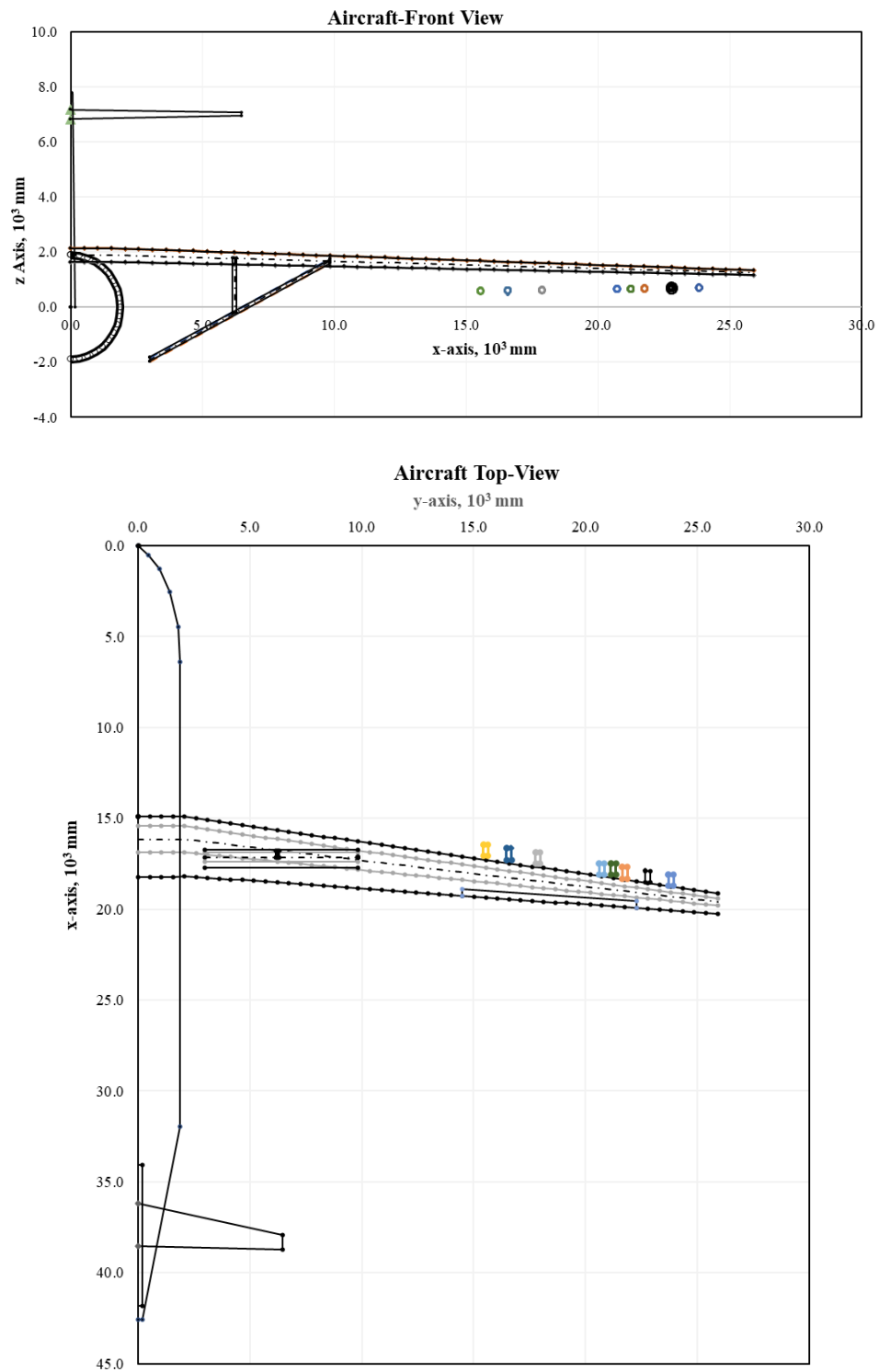


Figure 107 2D top and the front views of the optimised TBW with three optimisation variables.

10 DISCUSSION

This chapter discusses the advantages, disadvantages, limits, and potential of the research presented in this thesis by comparing it with the relevant studies in the literature. Before proceeding this section, the readers are recommended to review Sections - 1.7 and 1.14 as they contain supporting content that will enhance the understanding of the subsequent discussions.

10.1 Literature Review

A thorough exploration of strut- or truss-braced wing aircraft literature has been conducted. This review covers previous studies while emphasizing recent ones, providing a balanced review in the field since the last broad review by Cavallaro and Demasi in 2016. Moreover, another study conducted by Dababneh and Kipouros in 2018; focused on reviewing ME studies for conventional wings, but did not comprise the field of SBWs and TBWs. Consequently, the author undertook the task of filling this research gap by providing a comprehensive review of ME methods specifically tailored to SBWs and TBWs.

The studies have been categorised into subsections based on their respective sub-disciplines and objectives. This classification not only facilitates future researchers' access to relevant studies but also assists understanding the evolution and progression of the field.

A detailed understanding of strut-braced and truss-braced wing aircraft was attained through this review process. Recognizing the gaps, especially the lack of studies in the structural design and mass estimation (SDaME) of the SBW and TBW with conventional, hydrogen, electric and distributed propulsion systems at the conceptual and early preliminary design stage enabled the selection of those topics as the thesis focus. This gap identification eventually led to the decision to develop a mathematical model (MM) to study these areas.

The review process provided an understanding of the interconnection of various sub-disciplines, paving the way for a sophisticated MM. The model's compatibility with other sub-disciplines was a critical requirement and has been addressed effectively.

This literature review presents a foundation for future research in the field. Though it's a comprehensive review, certain additions such as figures and tables from previous studies could enhance its value further. While these additions have not been made

completely due to copyright concerns, the author has supplied essential illustrations pertaining to the thesis objectives.

In conclusion, this literature review can significantly contribute to future research, potentially accelerating developments in the field. With minor enhancements, it holds the potential for publication in reputable scientific journals.

10.2 Aircraft Geometric Model

The research has yielded modified geometric models aimed at facilitating detailed preliminary design stage studies of aircraft components, with a focus on wing and wing box structures. Following the outcomes of previous research (F. Gern *et al.*, 2000; Gern *et al.*, 2001; Sulaeman, Kapania and Haftka, 2002), a modified hexagonal wing box model has been introduced, which offers significant advantages for accurate flexible wing modelling, torsional stiffness computation, and ME.

Utilising a discrete geometric approach, the developed wing model allows unlimited definition of wing sections in the spanwise direction, a capability inspired by earlier studies (F. Gern *et al.*, 2000; Gern *et al.*, 2001; Locatelli *et al.*, 2016). This model accommodates variations in wing and wing box design parameters at each section, or, alternatively, enables computation through linear interpolation.

The new model significantly advances beyond existing conceptual or early preliminary wing and wing box models (Torenbeek, 1971; Macci, 1995; Gur *et al.*, 2011; Bhatia *et al.*, 2012; Zhang *et al.*, 2012; Chiozzotto, 2016, 2017) by providing a platform for studying the impact and optimisation of internal wing box components and wing geometries.

While primarily serving for SSaME of innovative wing configurations, the model's discrete architecture also caters to intermediate and low fidelity level aerodynamic and structural analyses (Song *et al.*, 2014; Cheung *et al.*, 2019; Nguyen *et al.*, 2019)

Furthermore, the model has proven to be instrumental in exploring distributed propulsion systems and dry wings for hydrogen and electric propulsion. In this respect, its application is foreseen for optimising wings for these developing technologies.

In addition, 3D geometric models have been developed for strut, jury, and offset, unparalleled in their detail compared to previous works (Park, 1980; Chiozzotto, 2016, 2017)

In conclusion, robust geometric models have been created, they meet to the thesis's objectives, focusing primarily on SSaME of innovative wings of varied configurations. These models provide a foundation for the structural design and sizing methods discussed in Chapter - 5 (STRUCTURAL ANALYSIS AND SIZING) and indicate an exciting opportunity for future studies.

10.3 Loading Actions

In Chapter - 4 (LOADING ACTIONS), several streamlined methods were developed for the analysis of truss component loads, complete aircraft mass moment of inertia calculation, and design and computation of wing fuel tank and fuel mass. Despite their simplicity and computational efficiency, these methods were cautiously selected, accounting for the sensitivity of the structural component mass to these load models.

A thorough literature review highlighted a lack of clear procedures for defining the number and types of critical loading cases necessary for the SSaME of both conventional and novel wings. As shown in Table 54, the range of loading cases varies across studies. To comprehensively address this, symmetric, rolling, and combined loading cases were covered following the airworthiness guidelines of EASA (2015, 2020). This approach resulted in a total of 483 full loading cases, the broadest range to date in initial aircraft design work.

Table 54 Number of loading cases used in the previous studies in the literature.

Reference Studies	Number of Loading Cases
(Riggins, 2015)	18
(Gupta <i>et al.</i> , 2014)	17
(Park, 1980), (Chiozzotto, 2016)	9
(Naghshineh-Pour, 1998), (F. Gern <i>et al.</i> , 2000), (Gur <i>et al.</i> , 2011)	3
(Elham, La Rocca and Vos, 2011), (Wang <i>et al.</i> , 2015), (Anil Variyar and Thomas D. Economon, 2016)	2
(Canosa, 2014)	1

The analysis of these cases revealed critical ones that directly influence structural sizing, and this will be discussed in the next sections. Furthermore, concentrated loads with their inertia and thrust were defined, allowing for any number of concentrated loads to be included in load analysis of the wing.

While the geometric model and structural sizing methods developed are compatible with intermediate fidelity level aerodynamic analyses like VLM, panel method, and inviscid

CFD analysis; simple lift distribution models were selected with user-defined aerodynamic constants at this stage of the project. The advantages of the simple semi-empirical lift distribution models are short computational, reduced complexity in their implication. These models are also compatible with the approaches utilised in this study such as considering the aeroelastic penalties with empirical methods.

Higher fidelity aerodynamic methods can be considered in the future steps of this study to increase the model sensitivity to wing planform shape, wing twist and to be used with aeroelastic models. The higher fidelity methods can also be used with the simple methods presented in this study to achieve a multi-fidelity level model which can achieve shorter computational time while taking the advantages of intermediate fidelity level aerodynamic methods. The future research can couple these aerodynamic tools with the developed MM for more detailed studies.

10.4 Structural Analysis and Sizing

The extensive review of wing box models in literature led to the creation of a procedure for sizing the structural elements of the modified, modular hexagonal wing box model, ensuring to meet yielding, failure stress, panel and column buckling (with plasticity effect) criteria. Bird strike is also considered to size the thickness fixed LE and TE. In addition to the detailed sizing constraints, detailed stress analyses are also conducted for each wing box skin panel, spar webs, caps and mass booms at each wing section.

The alternative method from Ref. (Megson, 2013) was adopted for computing shear flow on the skin panels of the idealised wing box, given its simplicity and completeness, it properly suits preliminary design studies, providing equations to size all wing box skin panels.

Many existing wing box models do not account for the impact of the wing box's internal structural design on its mass, due to the omission of detailed elements such as spar webs, spar caps, stringers, detailed skin panels, or detailed direct representation of wing box cross-sections (Bindolino *et al.*, 2010; Gur *et al.*, 2010; Elham, La Rocca and Vos, 2011; Elham and Van Tooren, 2014; Andrews, Perez and Wowk, 2015). In contrast, the proposed model facilitates an in-depth study of these factors. Additionally, this comprehensive sizing procedure allows us to design and optimise both conventional and dry wing boxes for conventional and distributed propulsion systems and reduce uncertainties in novel wing box mass estimation while enable us to study the effect those structural elements on the wing box mass.

In terms of the structural sizing of strut, jury, and offset, the proposed methods present one of the most detailed approaches, markedly addressing the previously unclear procedures for jury and offset sizing. The accuracy of these methods was validated in Chapter - 6 (MASS ESTIMATION and VALIDATION).

Dealing with strut-braced wing (SBW) configurations introduces indeterminate structural problems, typically addressed at early design stages by computing the strut reaction force applied to the wing. Existing solutions utilise Euler beam equation (Park, 1980), iteration (Locatelli *et al.*, 2016), and Castigliano's energy theorem (Canosa, 2014) based approaches. Additionally, Naghshineh-Pour (1998) defined this reaction force as a user input; however, this approach might not be suitable for design exploration and optimisation studies. The analytical models account for only a single concentrated load (Park, 1980; Canosa, 2014; Chiozzotto, 2016). One unique contribution of this study is the derivation of an analytical solution to compute the strut reaction force with unlimited concentrated loads using Castigliano's theorem.

This study has not covered the impact of flexible wings on lift distribution yet as it relies on simple aerodynamic lift distribution models. Future studies are suggested to employ a flexible wing model with higher fidelity aerodynamic methods, such as VLM and panel method.

10.5 Mass Estimation and Validation

Chapter - 6 (MASS ESTIMATION and VALIDATION) discloses an elaborate aircraft group mass statement, produced by critically reviewing numerous references and technical reports (Burt, 1962; Torenbeek, 1971; York and Labell, 1980; Macci, 1988; Howe, 1996, 2000; Chiozzotto, 2017; Raymer, 2018; Roskam, 2018), as shown in Table 1. This comprehensive mass statement promises to mitigate errors in future research.

The accuracy of the novel MM developed in this study was benchmarked against seven existing semi-empirical and quasi-analytical ME techniques (York and Labell, 1980; Jenkinson, Simpkin and Rhodes, 1999; Howe, 2000; Chiozzotto, 2017; Raymer, 2018). This comparison was performed across thirteen cantilever wings with metallic wing boxes and one TBW with quasi-isotropic composite wing box. The proposed method outperformed, evidenced by the lowest error indicators in wing group ME, namely standard deviation of 6.49%, standard error of 1.74%, and an average error of -2.22%. In terms of wing box, ME, the figures were 8.46%, 2.29%, and 1.07%, respectively.

Further comparison was conducted for increasing ARs from 5 to 40, incorporating sweep angle variations, and visualising wing group mass variation through surface plots. After AR of 10, the developed method showed more sensitivity to AR increases, while other models exhibited mostly a linear increase in estimated wing mass.

While the method's reaction to increasing AR aligns with wing ME expectations for UHARWs, additional validation is needed when more comprehensive UHARW mass data becomes accessible.

A detailed approach was introduced for the SSaME of truss components including strut, jury, and offset. This was validated against NASA's SUGAR Aircraft with TBW, yielding an error of -15.46% and -11.29% in strut and jury ME respectively, compared to data from a detailed design study (Bradley, Allen and Droney, 2015). This level of accuracy is unparalleled in the literature.

In conceptual or preliminary design studies, a trade-off exists between incorporating aeroelasticity and maintaining the model's capacity for detailed structural sizing. Aeroelasticity inclusion at an intermediate fidelity level necessitates significant geometric model simplifications, reducing accuracy and limiting design space exploration. For this reason, this study excluded aeroelasticity constraints within the physics-based MM and substituted them with simple empirical equations. These equations calculate mass penalties resulting from aeroelastic constraints for metallic and composite materials. The validity of the empirical approach in wing mass estimation was shown in Section 6.2.2.3. Future studies with higher fidelity aeroelastic models are recommended to evaluate and improve the accuracy of the proposed mass estimation method.

High-fidelity structural and aerodynamic analysis methods, despite their accuracy, often have extensive computational times, making them less desirable at the conceptual design stage (Werner-Westphal, Heinze and Horst, 2008; Cavagna, Ricci and Travaglini, 2011; Andrews, Perez and Wowk, 2015; Benaouali and Kachel, 2019). In contrast, the new method requires less than 250 *ms* for one loading case on an Intel i7-11800H CPU, an octa-core 2.3 GHz computer. This significant reduction in computational time, compared to the hours-long computations with higher fidelity tools, highlights the benefits of the new intermediate fidelity level physics-based approach and indicates its potential when implemented with high-level programming languages, rather than spreadsheet implementations.

10.6 Parametric and Sensitivity Analysis

In Chapter - 7 (PARAMETRIC and SENSITIVITY STUDIES), a thorough parametric analysis has been conducted for moderate to HAR cantilever, TBWs, and SBWs, using a one-factor-at-a-time approach, one of the most comprehensive studies of its kind in the literature. The sensitivity of wing box and wing group mass to design parameters aligns well with existing literature for cantilever wings. The truss components' mass variations also correspond with available literature, albeit limited.

With no apparent abnormalities, the results support the author's understanding and predictions. The outcomes confirm the tool's dependability and responsiveness to design constraints, supporting its application in next exploratory investigations.

The effect of the offset height on the mass of the wing group is one noteworthy finding. Despite the fact that an increase in offset height was predicted to cause the wing group mass to decrease for offset heights up to 1m, the results show the opposite. This unanticipated result, attributable to modifications to the load route and the ensuing decrease in wing box mass, has broad ramifications. The offset, which was originally designed to lessen the interface drag between the strut and the wing, now also promises to reduce the mass of the wing group, calling for additional investigation using high fidelity tools. This preliminary finding, which has received little attention in the literature, might have an impact on further studies and greatly improve the performance of TBW aircraft.

The optimal positioning of the strut attachment for minimum wing mass, as suggested by this research, coincides with other studies (Naghshineh-Pour, 1998; Gern *et al.*, 2001; Chiozzotto, 2016). Furthermore, the observed trends of wing box mass variation with thickness-to-chord ratio and AR values align with findings from (Olson and Henricks (2018), Solano *et al.* (2021), and Sarojini *et al.* (2022).

In conclusion, the insights gleaned from this detailed parametric study could guide future designers in developing a keen intuition for TBW aircraft and assist researchers in concentrating on various aspects of TBW aircraft. Nevertheless, these preliminary findings necessitate validation with higher fidelity tools.

10.7 Critical Loading Cases Investigation

This chapter addresses an often-overlooked aspect in conceptual and preliminary design stages – the feasibility of considering the complete loading cases as specified in

airworthiness requirements. While comprehensive coverage of these cases is computationally intensive, it was observed from literature review (Chapter - 2) that critical loading cases influencing structural sizing vary across different studies as previously listed in Table 54. Although the loading cases significantly impact the accuracy of component sizing, no study to date has examined their effects and systematically minimized their number, focusing solely on dominant cases without compromising accuracy in SSaME.

With a new approach, this study investigated the critical loading cases governing the structural sizing of novel wings. The use of only nine critical reduced loading cases led to a thirtyfold decrease in computational time while maintaining nearly the same level of accuracy as the complete loading cases.

This pioneering exploration presents future researchers with a potential roadmap to follow, minimizing the uncertainties often encountered in structural analysis studies. By focusing on the dominant loading cases, both accuracy and efficiency can be significantly improved in the early stages of design.

10.8 Comparative and Optimisation Studies

Chapter - 9, delivered several comparative and optimisation studies. While some of these findings are new and advance the discipline, others provide crucial comparison data to the body of already published work. A notable example is the association between the usage of a strut and ensuing decrease in wing mass which is in line with results from other research reviewed in Chapter - 2. It was also shown that using composite materials can result in further mass reductions for wings.

In particular, the design of dry HAR TBWs for hydrogen and electric aircraft as well as TBWs with distributed propulsion was examined. These are areas that have received little research. The originality of these study topics was highlighted in the literature review in Chapter - 2, and the findings are likely to motivate more studies. Due to the current lack of publicly available data from more in-depth investigations, it was unable to compare or confirm the methods for these combinations. Future studies are suggested to consider employing higher fidelity methods for validation and comparability.

But for the first time, the presented preliminary findings provide insight into how these cutting-edge technologies can affect the structural mass of the wing. The mass variation

of the wing components with different material types, hydrogen, electric or distributed propulsion technologies were shown.

Sample optimisation studies have been presented using the new MM. The optimisation results match with similar studies in the literature and the author's expectations. There was a literature gap in the optimisation of the TBW with distributed propulsion systems. It has been shown that the optimised TBW with distributed 16 engines were slightly heavier than the optimised TBW with two engines for the first time in the literature. This outcome should be verified with higher fidelity tools. The suitability of the new MM model in optimisation studies was also presented.

Even though this work investigated the effect of distributed propulsions on wing group mass, the use of crude aerodynamic models must be acknowledged. The present model ignores the important impact that distributed propulsion systems have on the aerodynamics of the wing. Assuming a homogeneous mass distribution across each engine with a total mass equal to that of typical propulsion systems, the propulsion sizing technique was also relatively basic. Although these presumptions worked well to show the capabilities of the new MM and the structural response of wings to these technologies, future research should make use of more advanced aerodynamic and propulsion design models together with the new MM.

10.9 Summary

10.9.1 Contribution

This thesis offers significant contributions to the existing body of knowledge in aeronautical engineering, focusing on the structural design, sizing, and ME of moderate to HAR SBW and TBW aircraft with novel technologies. It introduces a new, precise MM for the task, which provides higher accuracy, rapid computational time and lower standard error compared to existing semi-empirical and quasi-analytical ME methods. The research also investigates the critical loading cases in aircraft design, substantially reducing computational time without sacrificing accuracy. Lastly, it presents one of the first studies examining the impact of novel propulsion technologies on the structural mass of studied wing designs.

10.9.2 Contextualisation

The work contextualises within the evolving landscape of aircraft design, which has been increasingly seeking more efficient, economical, and environmentally friendly solutions.

High and UHARWs and TBWs represent promising avenues in achieving these goals. The research contributes to reducing uncertainties in the SSaME of these innovative designs. Additionally, it paves the way for further investigations into the implications of emerging propulsion technologies in aircraft design.

10.9.3 Limitations

The primary drawbacks of this research are its dependence on low fidelity aerodynamic and aeroelasticity methods in the MM. The findings may not fully account for all potential design factors even though the model described performs well and speeds up calculation. This is due to the absence of comparison with higher fidelity approaches in some design instances as UHARWs. Similar to how the effects of distributed propulsions was examined, the impacts of the spanwise lift distribution may not have been entirely captured by the basic aerodynamic models used. The study recognises these limitations and suggests more validation and investigation utilising better quality tools and methodologies.

11 CONCLUSION

Approaching the conclusion of this doctoral research, it is appropriate to reflect on the studies that have included the thorough investigation of wing SSaME for novel aircraft configurations by developing an innovative physics-based MM, a thorough validation against existing aircraft, research and design studies with this MM.

To identify gaps in the current SSaME approaches and other disciplines, a systematic review of the literature on both strut- and truss-braced wing aircraft was conducted as the first step in the research. These gaps were primarily caused by traditional methodologies' inability to address the structural considerations of wings accurately and sensitively with moderate to HARs, regardless of whether they are cantilever, strut-, or truss braced, and in the context of developing propulsion technologies.

An advanced, particularly sensitive, computationally quick, and physics-based strut, jury, offset, wing box, and wing group mass estimate approaches were developed in the thesis to study the literature gaps. Following a thorough validation of this method using information from thirteen real aluminium alloy wing boxes and wing groups as well as one composite HAR TBW, the standard errors for the mass of the wing box and the mass of the wing group, respectively, were found to be 2.29% and -1.07%. The high level of accuracy in the jury and strut ME is unparalleled in the literature when taking into account the same fidelity level techniques.

The novel geometric models and detailed loading analysis approach presented in this thesis were crucial components to achieve the accuracy and computational efficiency of the developed method. The adaptability of these models to represent various internal structural elements of wing boxes and various wing planform types, as well as their capacity to define a variable number of wing sections and simulate a wide range of loading cases, all attest to the robustness and versatility of the developed methodology.

Another key contribution of this thesis was the demonstration of the methodology's effectiveness in providing insights into the impact of various aircraft design parameters on the strut, jury, offset, wing box and wing group masses. The extensive parametric tests that were conducted for various aircraft configurations highlighted this. Additionally, the mass variations of the wing box, wing group, and other components in moderate to HAR, cantilever, or TBWs with hydrogen, electric, or distributed propulsion systems were disclosed through parametric studies.

A unique facet of this study was the focus on critical loading cases in the structural sizing of innovative wings, an aspect often overlooked during the conceptual and preliminary design stages. By centring on only nine dominant loading cases, a significant reduction in computational time was achieved without compromising accuracy, establishing a roadmap for future research and offering a way to navigate the uncertainties often faced in structural analysis studies.

Another milestone was the comparative and optimization studies carried out. The research added crucial comparative data to the body of work in the field and identified new insights, particularly concerning the use of a strut and the resultant decrease in wing mass. Additionally, it explored less researched areas like the design of dry HAR TBWs for hydrogen and electric aircraft and TBWs with distributed propulsion, contributing preliminary insights about these technologies' effect on the wing's structural mass.

Despite the substantial accomplishments of this research, it is important to note that it was carried out within certain constraints. For instance, the lift distributions in the current mathematical model are based on straightforward aerodynamic models, and the aerodynamic coefficients are currently provided by the user. Additionally, the implications of composite wing box design using optimisation techniques and flexible wing model effects were outside the purview of the current thesis.

Nonetheless, the potential for future research does not diminish the value of the work presented in this thesis. On the contrary, it speaks to the versatility and flexibility of the methodology developed in this research, as it is amenable to incorporation with additional models, parameters, and future technologies.

In summary, this doctoral research has significantly contributed to the field of aircraft design, particularly in the conceptual design phase. It has introduced novel, physics-based wing box and wing group ME methodologies, which hold high relevance for novel aircraft configurations and pave the way for future advancements in the field. The findings also highlight the importance of continuous research in this area to enhance the efficiency and performance of future aircraft designs.

12 FUTURE WORK

This chapter presents the future directions of this research. Section - 12.1 outlines the academic articles that could be derived directly from this thesis with minimal additional work. Each potential publication is assessed by completion status, feasibility, and projected impact. Following, Section - 12.2 explores the long-term evolution of this project, envisioning how the methodologies presented in this thesis can be expanded and refined in future research efforts. This chapter provides a forward-looking roadmap for immediate scholarly outputs and long-term contributions to aircraft design and ME.

12.1 Potential Publications Arising from this Thesis

The thorough and interdisciplinary nature of the research done for this thesis has the potential to produce several unique papers that will add to the body of knowledge on aircraft design and ME. The possible publications listed below have been identified, together with information on their probabilities of completion, feasibility given the work already done, and potential effects:

1. "*Development of a Design Sensitive Wing Mass Estimation Method: Application with Distributed Propulsion and Emerging Fuel Systems.*" This article (90% complete) will elaborate on the development and implementation of proposed ME methods for metallic and composite cantilever wings and their application with emerging propulsion technologies. This paper's feasibility is high and is derived directly from primary research focus of this study. Potential impact includes offering a crucial tool for the conceptual design of future aircraft and fostering the transition to more sustainable propulsion technologies.
2. "*Structural Sizing and Mass Estimation of Strut- and Truss-braced Wings with Emerging Propulsion and Fuel Technologies.*" Building upon the previous paper, this second article (90% complete) extends the ME method to strut- or truss-braced wings. The feasibility of this extension is high, based on the foundation provided by the initial method development. This work has a significant potential impact, providing a much-needed resource for design studies of novel aircraft configurations.
3. "*Exploring Structural Design and Weight Considerations for Strut-Braced and Cantilever Wing-Boxes.*" This paper (90% complete) presents a detailed parametric analysis of strut-braced and cantilever wing boxes, offering critical design insights. Its feasibility is ensured by the depth of analysis conducted. The

potential impact lies in informing and guiding the aviation design community towards more efficient and effective wing box designs.

4. *"Critical Loading Scenarios for Mass Estimation and Structural Design of Cantilever, Strut- and Truss-Braced Wings with Emerging Fuel and Propulsion Technologies."* This article (90% complete) focuses on the role of various loading cases in the structural design and ME (SDaME) processes. Its feasibility is rooted in the integral role loading scenarios play in the proposed model. The potential impact is considerable, helping designers better understand and plan for the forces these innovative wing designs may experience.
5. *"Optimizing Strut- and Truss-Braced Wings with Distributed, Hydrogen, and Electric Propulsion: A Structural Design Perspective."* This article (70% complete) integrates various aspects of the research to discuss the optimization of strut- and truss-braced wings, particularly for emerging propulsion technologies. Its feasibility is guaranteed by the interdisciplinary nature of the work. The potential impact is significant, contributing to the sustainability discourse in aviation.
6. *"A Review and Assessment of Strut- and Truss-Braced Wing Aircraft: Potentials, Challenges, and Solutions"* This review article (75% complete) consolidates the current research and practice in strut- or truss-braced wing aircraft. It builds upon the extensive literature review, ensuring its feasibility. The potential impact is substantial, offering a consolidated source of existing knowledge for other researchers in the field.

Each potential publication stems directly from the research conducted in this thesis and aligns closely with its overall aims and objectives. These papers, once published, will make a substantial contribution to the academic and industrial understanding of design methodologies for innovative aircraft configurations.

12.2 Next Steps of this Project

There are several intriguing directions for future study that, when built upon, have the potential to deepen and broaden the understanding of the SDaME of unique aircraft configurations:

Semi-empirical models: Mass data of the investigated wing configurations may be produced with the created intermediate fidelity level mathematical model (MM) and sampling methodologies. This dataset can then be used in regression analysis to create

even more straightforward semi-empirical mass estimate models for each wing component, decrease complexity, and speed up processing.

Integration with Aerodynamic Models: The MM developed in this thesis relies on simple aerodynamic models to compute lift distributions, and the aerodynamic coefficients are user input at this stage. Future work could focus on the integration of this model with intermediate fidelity level aerodynamic tools to enable more precise simulations of wing behaviour under various load cases.

Aeroelastic Effects: This research has primarily centred around rigid wing models and the aeroelastic effects are only covered with existing empirical methods. The aeroelastic effect is an important aspect especially in high and UHARWs that would be an interesting exploration in the context of this work and can be considered in future studies.

Optimization of Composite Wing Box Design: Although the detailed composite wing box design using optimization methods has been discussed, the effects of these optimized composite laminates on the final wing mass have not been studied in this thesis. Future work could involve conducting these studies using the developed composite analysis and optimization tools based on classical laminate theory and macro mechanics of composite materials.

Extension to Other Aircraft Design Parameters: In this thesis, the effect of some aircraft design parameters on the wing box mass was explored. Further research could delve into the effects of additional aircraft design parameters using the same research method, thus enhancing the scope and applicability of this research.

Further Validation Studies: There is opportunity for additional validation experiments using higher-order WB ME methods especially for UHARWs, even though the novel wing and wing box ME methodologies have been validated for moderate to HAR cantilever and SBWs. This might both reveal potential areas for development and increase credibility of the present MM.

Broad Application of the Mathematical Model: The full potential of the developed MM has not been fully tested in this work. Therefore, its application for more detailed optimization studies can open new paths for contributions to the field of aircraft design. Potential applications could include the single or multidisciplinary aircraft design and optimisation with hydrogen or electric propulsion, distributed propulsion, or UHARWs.

In conclusion, there is a promising scope for the further development and application of the research presented in this thesis. As the aerospace industry continues to evolve and face new challenges and opportunities, the tools and techniques developed here will remain relevant, adaptable, and valuable in the conceptual design and optimisation of novel aircraft configurations.

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APPENDIXES

Appendix A Loading Actions

A.1 Rolling Case Calculations

A.1.1 Airworthiness Requirements

The following requirements are followed while generating the rolling cases:

1. CS.25.349(a) is utilised for the required roll case.
2. AMC CS25.147 is utilised for computing roll timing.
3. CS.25.349(b) is utilised for gust alleviation capability.

A.1.2 Roll Phases

The following three roll phases are covered in the computations.

1. Roll initiation
2. Steady roll
3. Roll arrest

A.1.3 Sources of Loads

The following three sources of loads are considered.

1. Aerodynamic Loads
2. Damping Loads
3. Inertia

A.1.4 Computation of Rolling Loads

The following steps are followed to compute the rolling loads.

1. Computing required p .
2. Compute required ξ_{HS} and ξ_{LS} to achieve the required p .
3. Verify that these aileron deflections meet the roll timing requirement; if not, update ξ_{HS} and ξ_{LS} .
4. Compute the aileron aerodynamic load and $\dot{p}$.
5. Compute damping and inertia loads.
6. Compute the total rolling load for each roll phase.

in which,

p – Angular rate of roll (Roll velocity).

ξ_{HS}, ξ_{LS} – Aileron deflection. HS and LS are for high speed and low speed, respectively.

$\dot{p}$ – Roll acceleration

L – Rolling moment

$\bar{L}$ – Non-dimensional rolling moment

b – wing-span

A.1.5 Total Load Computation

The rolling loads are combined with the relevant symmetric manoeuvre loads, as per CS.25.349(a).

A.1.6 Load Distribution of the Aileron

The spanwise load distribution of the aileron is assumed as a rectangular shape as the default of the method. The effect of the additional lift in torsion can be taken into account with the wing section pitching moment coefficient.

A.2 Loading Calculations

The rolling load calculations are given in Section 4.5.2. This section provides detailed derivations for the computation of the terms.

A.2.1 Roll Rate and Acceleration

From lecture notes of Cranfield University (Refs. DAeT9585, and DAeT9586), the moment equation for the roll is,

$$L_{\xi} + L_{\zeta} + L_P + L_V = I_{xx} \times \dot{p} - I_{xy} \times \dot{r} \quad (\text{B-1})$$

I_{xy} can be neglected for high aspect ratio wings. The roll and yaw kinematics are considered entirely independent in the early design stage. Thus, the terms L_r, L_{ζ}, L_v will be neglected. Hence,

$$L_{\xi} + L_P = I_{xx} \times \dot{p} \quad (\text{B-2})$$

The non-dimensional form of the terms in B-2 are,

$$\overline{L}_\xi = L_\xi / (0.5\rho SV^2 b) \quad (\text{B-3})$$

$$\overline{L}_p = L_p / (0.5\rho SV b^2) \quad (\text{B-4})$$

If the Eq. (B-3) and (B-4) are substituted in Eq. ((B-2),

$$(0.5\rho SV^2 b)[\overline{L}_\xi \xi] + (0.5\rho SV b^2)[\overline{L}_p p] = I_{xx} \times \dot{p} \quad (\text{B-5})$$

ξ can be calculated for HS and LS,

$$[\overline{L}_\xi \xi] = \overline{L}_\xi^{LS} \xi_{LS} + \overline{L}_\xi^{HS} \xi_{HS} \quad (\text{B-6})$$

Equation (B-5) can be rearranged as,

$$\frac{I_{xx} \times \dot{p}}{0.5\rho SV b^2} - \frac{pb}{V} \overline{L}_p = \overline{L}_\xi^{LS} \xi_{LS} + \overline{L}_\xi^{HS} \xi_{HS} \quad (\text{B-7})$$

or, more conveniently as,

$$\dot{p} + K_1 p = K_2 \xi_{LS} + K_3 \xi_{HS} \quad (\text{B-8})$$

where,

$$K_1 = \frac{-0.5\rho SV b^2}{I_{xx}} \overline{L}_p \quad (\text{B-9})$$

$$K_2 = \frac{0.5\rho SV^2 b}{I_{xx}} \overline{L}_\xi^{HS} \quad (\text{B-10})$$

$$K_3 = \frac{0.5\rho SV^2 b}{I_{xx}} \overline{L}_\xi^{LS} \quad (\text{B-11})$$

For the specific case of a sudden aileron deflection (as presented by CS.25.349(a)), (B-8) yields the solution.

$$p = \left[\frac{K_2}{K_1} \xi_{HS} + \frac{K_3}{K_1} \xi_{LS} \right] (1 - e^{-K_1 t}) \quad (\text{B-12})$$

A.2.2 Roll Phases

According to CS25.349(a), the rolling manoeuvre can be analysed in three discrete phases,

Roll Initiation:

- i. As the roll rate is zero, no damping loads are present
- ii. Max roll acceleration
- iii. Sudden deflection from zero to max

$$p = 0 \quad (\text{B-13})$$

$$\dot{p} = K_2 \xi_{HS} + K_3 \xi_{LS} \quad (\text{B-14})$$

Steady Roll:

- i. The damping moment is equal to the aileron moment
- ii. Max roll rate
- iii. No roll acceleration and no inertia loads

$$p = \frac{1}{K_1} [K_2 \xi_{HS} + K_3 \xi_{LS}] \quad (\text{B-15})$$

$$\dot{p} = 0 \quad (\text{B-16})$$

Roll Arrest:

- i. Ailerons are suddenly brought to zero
- ii. The roll rate is at its maximum
- iii. No aileron loads are present

$$p = \frac{1}{K_1} [K_2 \xi_{HS} + K_3 \xi_{LS}] \quad (\text{B-17})$$

$$\dot{p} = -K_1 p \quad (\text{B-18})$$

A.2.3 Aileron Deflection Angles

ξ_{HS} and ξ_{LS} must be determined to obtain p and $\dot{p}$. These are constrained by two requirements CS.25.349 and AMC CS.25.147.

Compliance with CS.25.349:

1. At V_A , a sudden maximum aileron deflection is assumed.
2. At V_C , the necessary aileron deflection to obtain the same roll rate as at V_A is assumed.
3. At V_D , the necessary aileron deflection to obtain a roll rate of 1/3 of that at V_A is assumed.

These can be concisely written as,

$$P_{req \text{ at } V_A} = P_{V_A}$$

$$P_{req \text{ at } V_C} = P_{V_A}$$

$$P_{req \text{ at } V_D} = 1/3 P_{V_A}$$

For aircraft with two sets of ailerons, at this stage, we may assume a full deflection of the high-speed aileron and compute the necessary low-speed aileron deflection to meet the required roll rate.

Compliance with AMC.25.147

As AMC.25.147 is complicated, it is not considered yet. Hence only AMC.25.147 is considered here. The aircraft must be able to roll from -30° to 30° in 7 seconds. Considering Eq.(B-12) with $p = \frac{d\phi}{dt}$ and a newly defined parameter of $\lambda = \frac{K_2}{K_1} \xi_{HS} + \frac{K_3}{K_1} \xi_{LS}$, the following equation can be written.

$$\frac{d\phi}{dt} = \lambda(1 - e^{-K_1 t}) \quad \text{(B-19)}$$

$$\phi(t) = \lambda \left(t + \frac{e^{-K_1 t}}{K_1} + \phi(t_0) \right) \quad \text{(B-20)}$$

The necessary λ parameter to comply with the AMC.25.147 requirement is, therefore,

$$\lambda_{req} = \left(\frac{\phi(t_{roll}) - \phi(t_0)}{t_{roll} + \frac{e^{-K_1 t_{roll}} - 1}{K_1}} + \right) \quad \text{(B-21)}$$

In which,

$$t_{roll} = 7 \text{ seconds}$$

$$\phi(t_0) = -\frac{\pi}{6} = -30^\circ$$

$$\phi(t_{roll}) = +\frac{\pi}{6} = +30^\circ$$

When $\xi_{HS} = \xi_{max}$, ξ_{LS} can be found as

$$\xi_{LS,req} = \frac{K_1}{K_3} \left[\lambda_{req} - \frac{K_1}{K_2} \right]$$

If $\xi_{LS,req,AMC} > \xi_{LS,CS}$ then $\xi_{LS,CS}$ need to be updated.

Appendix B Aircraft Data for Baselines and Validation

B.1 Aircraft Design Parameters

Table 55 Dataset-1 of aircraft used for validation.

No	Parameters (SI)	RB-66B	C-5A	C-9A	C-123B	C-133B	C-135B	C-141A	DC-8
1	Wing Root Chord (mm)	4,917.44	12,298.68	5,430.52	4,559.30	7,388.86	8,585.20	8,950.96	9,672.32
2	Wing Tip Chord (mm)	1,648.46	4,673.60	1,104.90	2,407.92	1,686.56	2,844.80	3,370.58	2,222.50
3	Default $\varepsilon_{fs,r}$	0.15	0.15	0.15	0.15	0.15	0.15	0.15	0.15
4	Default $\varepsilon_{fs,t}$	0.15	0.15	0.15	0.15	0.15	0.15	0.15	0.15
5	Default $\varepsilon_{rs,r}$	0.65	0.65	0.65	0.65	0.65	0.65	0.65	0.65
6	Default $\varepsilon_{rs,t}$	0.65	0.65	0.65	0.65	0.65	0.65	0.65	0.65
7	$\tau(y_r)$	0.10	0.12	0.13	0.17	0.17	0.17	0.12	0.12
8	$\tau(y_t)$	0.08	0.11	0.09	0.09	0.15	0.09	0.10	0.11
9	Default $\Psi_{wb,Edge,Root}$	0.70	0.70	0.70	0.70	0.70	0.70	0.70	0.70
10	Default $\Psi_{wb,Edge,Tip}$	0.70	0.70	0.70	0.70	0.70	0.70	0.70	0.70
11	Default $\Psi_{u,r}$	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
12	Default $\Psi_{u,t}$	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
13	LE Sweep Angle (deg.)	36.64	24.90	25.59	2.81	4.40	35.82	23.48	31.00
14	Default β_{Root}	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
15	Stringer Counts on One Surface	18.00	47.00	21.00	17.00	29.00	33.00	34.00	37.00
16	Wingspan (mm)	22,098.00	67,878.96	28,437.84	33,528.00	54,742.08	39,867.84	48,676.56	43,403.52
17	Wing Configuration	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever
18	Material Type	7075-T6	7075-T6	7075-T6	7075-T6	7075-T6	7075-T6	7075-T6	7075-T6
19	Default Plasticity Factor,	0.70	0.70	0.70	0.70	0.70	0.70	0.70	0.70
20	Wing Mount Type	High	High	Low	High	High	Low	High	Low

No	Parameters (SI)	RB-66B	C-5A	C-9A	C-123B	C-133B	C-135B	C-141A	DC-8
21	M_Wing_Group (kg)	3,991.61	37,648.17	5,216.31	2,676.19	12,201.63	11,566.61	15,648.94	15,195.34
22	M_Wing_Box (kg)	2,585.48	23,351.39	2,766.91	1,950.45	7,484.27	7,711.07	10,205.83	9,071.85
23	Calculated M_Wing_Fuel (kg)	4,354.49	142,428.00	8,346.10	3,991.61	81,646.63	45,359.24	70,306.82	65,770.89
24	Approx. V_Fuel_Total (m3)	15.00	170.00	13.90	10.00	45.00	20.00	66.00	66.40
25	TOGW (kg)	31,751.47	330,215.25	49,441.57	24,493.99	129,727.42	125,871.88	144,242.37	144,242.37
26	OEW (kg)	22,648.77	207,054.94	33,978.60	12,612.59	71,649.90	58,481.21	59,470.50	45,288.02
27	V_L_ or V_D (m/s)	303.01	210.92	205.78	102.89	141.47	185.20	210.92	231.50
28	M_D, EAS	0.88	0.71	0.93	0.40	0.51	0.93	0.80	0.87
29	V_C (m/s)	288.01	195.92	190.78	87.89	126.47	170.20	195.92	216.50
30	M_C	0.83	0.66	0.88	0.35	0.46	0.88	0.75	0.82
31	S_ROLL_Estimated (m2)	3.03	23.49	3.53	7.74	13.41	11.11	15.90	15.01
32	S_W_Estimated (m2)	72.46	576.00	92.97	113.64	248.34	226.03	299.90	257.57
33	S_FLAP_Estimated (m2)	10.03	92.13	19.58	11.89	46.13	33.63	49.12	42.45
34	S_LED_Estimated (m2) or S_SLAT or S_LEF	6.87	60.26	14.86	0.00	0.00	2.42	0.00	2.32
35	S_SPOIL_Estimated (m2)	1.58	40.01	3.86	0.00	0.00	9.96	28.89	4.51
36	Fuselage Length, mm	22,910.00	75,300.00	36,350.00	36,000.00	47,000.00	41,530.00	51,300.00	45,900.00
37	Fuselage Diameter (mm)	1,700.00	5,650.00	3,300.00	4,000.00	4,500.00	4,500.00	4,500.00	3,730.00
38	LG Mount Place	Fuselage	Fuselage	Wing	Fuselage	Fuselage	Wing	Fuselage	Wing
39	Approx. Λ _LG	0.15	0.15	0.18	0.18	0.18	0.18	0.15	0.10
40	Estimated Main LG Weight (kg)	1,100.19	5,646.68	845.45	418.85	2,218.34	2,152.41	2,466.54	2,466.54
41	Drop Tank Presence	Absent	Absent	Absent	Present	Absent	Absent	Absent	Absent
42	Approx. Λ _Drop_Tank	0.00	0.00	0.00	0.50	0.00	0.00	0.00	0.00
43	Approx. Ψ _Drop_Tank	0.00	0.00	0.00	-2.00	0.00	0.00	0.00	0.00
44	Approx. ε _Drop_Tank	0.00	0.00	0.00	0.30	0.00	0.00	0.00	0.00
45	DT Fuel Weight Estimated (kg)	0.00	0.00	0.00	1,000.00	0.00	0.00	0.00	0.00

No	Parameters (SI)	RB-66B	C-5A	C-9A	C-123B	C-133B	C-135B	C-141A	DC-8
46	Default $\Lambda_{\text{Aileron1_in}}$	0.55	0.55	0.55	0.55	0.55	0.55	0.55	0.55
47	Default $\Lambda_{\text{Aileron1_out}}$	0.85	0.85	0.85	0.85	0.85	0.85	0.85	0.85
48	Service Ceiling (m)	12,000.00	12,000.00	11,000.00	6,400.00	9,130.00	15,200.00	0.00	10,668.00
49	Ferry Range (km)	3,974.00	13,000.00	4,670.00	1,666.00	24,000.00	5,500.00	0.00	7,500.00
50	Airworthiness Requirements	CS25	CS25	CS25	CS25	CS25	CS25	CS25	CS25
51	Vertical Tail Position	Mid-Fin	Fin Top	Fin Top	Fuselage	Fuselage	Fuselage	Fin Top	Fuselage
52	Aspect Ratio Calculated	6.74	8.00	8.70	9.89	12.07	7.03	7.90	7.31
53	Nacelle Type	W Pylon	W Pylon	W Pylon	W/O Pylon	W/O Pylon	W Pylon	W Pylon	W Pylon
54	Engine Count on One Wing	1.00	2.00	1.00	1.00	2.00	2.00	2.00	2.00
55	Engine Mount Position	Wing	Wing	Fuselage	Wing	Wing	Wing	Wing	Wing
56	Estimated M_{Engine_1} (kg)	2,553.00	4,174.50	1,672.10	1,233.95	1,351.25	2,402.35	2,402.35	2,699.05
57	Approx. $\Lambda_{\text{Engine}_1}$	0.35	0.30	0.00	0.20	0.25	0.45	0.30	0.30
58	Approx. Ψ_{Engine_1}	-3.00	-3.00	-3.00	-3.00	-3.00	-3.00	-3.00	-3.00
59	Approx. $\epsilon_{\text{Engine}_1}$	-0.25	-0.25	-0.25	-0.25	-0.25	-0.40	-0.25	-0.25
60	Approx. Engine 1 Length (mm)	3,500.00	7,900.00	3,600.00	2,068.00	3,900.00	5,000.00	3,600.00	6,197.00
61	Approx. Engine 1 Diameter (mm)	750.00	2,460.00	1,300.00	1,342.00	857.00	1,300.00	1,300.00	990.00
62	Estimated M_{Engine_2} (kg)	0.00	4,174.50	0.00	0.00	1,351.25	2,402.35	2,402.35	2,699.05
63	Estimated $\Lambda_{\text{Engine}_2}$	0.00	0.55	0.00	0.00	0.50	0.73	0.50	0.55
64	Approx. Ψ_{Engine_2}	0.00	-3.00	0.00	0.00	-3.00	-3.00	-3.00	-3.00
65	Approx. $\epsilon_{\text{Engine}_2}$	0.00	-0.25	0.00	0.00	-0.25	-0.40	-0.25	-0.25
66	Approx. Engine 2 Length (mm)	0.00	7,900.00	0.00	0.00	3,900.00	5,000.00	3,610.00	6,197.00
67	Approx. Engine 2 Diameter (mm)	0.00	2,460.00	0.00	0.00	857.00	1,300.00	1,300.00	990.00

Table 56 Dataset-2 of aircraft used for validation.

No	Parameters (SI)	720	727	737	747	G-159	G-1159
1	Wing Root Chord (mm)	8,534.40	7,241.54	5,641.34	14,137.64	3,398.52	5,080.00
2	Wing Tip Chord (mm)	2,844.80	2,326.64	1,607.82	4,084.32	1,346.20	1,930.40
3	Default $\varepsilon_{fs,r}$	0.15	0.15	0.15	0.15	0.15	0.15
4	Default $\varepsilon_{fs,t}$	0.15	0.15	0.15	0.15	0.15	0.15
5	Default $\varepsilon_{rs,r}$	0.65	0.65	0.65	0.65	0.65	0.65
6	Default $\varepsilon_{rs,t}$	0.65	0.65	0.65	0.65	0.65	0.65
7	$\tau(y_r)$	0.16	0.15	0.11	0.10	0.14	0.11
8	$\tau(y_t)$	0.09	0.09	0.11	0.08	0.14	0.08
9	Default $\Psi_{wb,Edge,Root}$	0.70	0.70	0.70	0.70	0.70	0.70
10	Default $\Psi_{wb,Edge,Tip}$	0.70	0.70	0.70	0.70	0.70	0.70
11	Default $\Psi_{u,r}$	0.00	0.00	0.00	0.00	0.00	0.00
12	Default $\Psi_{u,t}$	0.00	0.00	0.00	0.00	0.00	0.00
13	LE Sweep Angle (deg.)	35.77	32.81	26.30	38.20	2.23	26.14
14	Default β_{Root}	0.00	0.00	0.00	0.00	0.00	0.00
15	Stringer Counts on One Surface	32.00	28.00	21.00	53.00	33.00	25.00
16	Wingspan (mm)	39,867.84	32,918.40	28,346.40	59,649.36	23,865.84	21,000.72
17	Wing Configuration	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever	Cantilever
18	Material Type	7075-T6	7178-T651	7075-T6	7075-T6	7075-T6	7075-T651
19	Default Plasticity Factor,	0.70	0.70	0.70	0.70	0.70	0.70
20	Wing Mount Type	Low	Low	Low	Low	Low	Low
21	M_Wing_Group (kg)	10,659.42	8,073.94	4,853.44	39,916.13	1,655.61	2,857.63
22	M_Wing_Box (kg)	6,803.89	4,082.33	2,585.48	25,854.77	1,043.26	1,814.37
23	Calculated M_Wing_Fuel (kg)	30,390.69	21,772.43	14,514.96	142,428.00	4,762.72	9,162.57
24	Approx. V_Fuel_Total (m3)	60.90	27.20	22.60	183.00	7.00	15.00
25	TOGW (kg)	92,079.25	73,028.37	45,722.11	322,957.77	15,921.09	25,627.97

No	Parameters (SI)	720	727	737	747	G-159	G-1159
26	OEW (kg)	36,513.28	41,761.80	27,026.39	165,646.04	6,615.64	13,588.72
27	V_L_ or V_D (m/s)	185.20	200.63	216.07	228.93	175.43	214.01
28	M_D, EAS	0.95	0.92	0.84	0.92	0.75	0.85
29	V_C (m/s)	170.20	185.63	201.07	213.93	160.43	199.01
30	M_C	0.90	0.87	0.73	0.87	0.70	0.80
31	S_ROLL_Estimated (m2)	11.11	5.30	2.50	20.62	3.40	2.68
32	S_W_Estimated (m2)	226.03	157.47	102.75	543.39	56.64	73.72
33	S_FLAP_Estimated (m2)	33.59	26.11	16.75	78.69	10.29	11.97
34	S_LED_Estimated (m2) or S_SLAT or S_LEF	8.86	18.58	9.35	41.62	0.00	0.00
35	S_SPOIL_Estimated (m2)	9.66	10.64	6.57	28.24	0.00	4.59
36	Fuselage Length, mm	41,500.00	40,590.00	31,200.00	76,300.00	23,930.00	24,360.00
37	Fuselage Diameter (mm)	3,760.00	3,800.00	3,760.00	5,600.00	2,800.00	2,800.00
38	LG Mount Place	Wing	Wing	Wing	Wing	Wing	Wing
39	Approx. Λ _LG	0.15	0.15	0.18	0.15	0.30	0.15
40	Estimated Main LG Weight (kg)	1,574.56	1,248.79	781.85	5,522.58	272.25	438.24
41	Drop Tank Presence	Absent	Absent	Absent	Absent	Absent	Absent
42	Approx. Λ _Drop_Tank	0.00	0.00	0.00	0.00	0.00	0.00
43	Approx. Ψ _Drop_Tank	0.00	0.00	0.00	0.00	0.00	0.00
44	Approx. ϵ _Drop_Tank	0.00	0.00	0.00	0.00	0.00	0.00
45	DT Fuel Weight Estimated (kg)	0.00	0.00	0.00	0.00	0.00	0.00
46	Default Λ _Aileron1_in	0.55	0.55	0.55	0.55	0.55	0.55
47	Default Λ _Aileron1_out	0.85	0.85	0.85	0.85	0.85	0.85
48	Service Ceiling (m)	12,802.00	10,730.00	11,300.00	13,715.00	10,200.00	14,000.00
49	Ferry Range (km)	5,200.00	4,170.00	2,850.00	10,800.00	4,090.00	6,570.00
50	Airworthiness Requirements	CS25	CS25	CS25	CS25	CS25	CS25
51	Vertical Tail Position	Fuselage	Fin Top	Fuselage	Fuselage	Fuselage	Fin Top

No	Parameters (SI)	720	727	737	747	G-159	G-1159
52	Aspect Ratio Calculated	7.03	6.88	7.82	6.55	10.06	5.98
53	Nacelle Type	W Pylon	W Pylon	W Pylon	W Pylon	W/O Pylon	W Pylon
54	Engine Count on One Wing	2.00	1.00	1.00	2.00	1.00	1.00
55	Engine Mount Position	Wing	Fuselage	Wing	Wing	Wing	Fuselage
56	Estimated M_Engine_1 (kg)	4,945.00	4,025.00	4,025.00	6,222.65	629.05	1,035.00
57	Approx. $\Lambda_{\text{Engine},1}$	0.40	0.00	0.25	0.25	0.30	0.00
58	Approx. $\Psi_{\text{Engine},1}$	-3.00	0.00	-3.00	-3.00	0.00	0.00
59	Approx. $\varepsilon_{\text{Engine},1}$	-0.50	0.00	0.40	-0.25	-0.25	0.00
60	Approx. Engine 1 Length (mm)	3,937.00	3,937.00	5,500.00	5,190.00	2,480.00	2,480.00
61	Approx. Engine 1 Diameter (mm)	990.00	990.00	1,250.00	2,190.00	960.00	960.00
62	Estimated M_Engine_2 (kg)	4,945.00	0.00	0.00	6,222.65	0.00	0.00
63	Estimated $\Lambda_{\text{Engine},2}$	0.70	0.00	0.00	0.55	0.00	0.00
64	Approx. $\Psi_{\text{Engine},2}$	-3.00	0.00	0.00	-3.00	0.00	0.00
65	Approx. $\varepsilon_{\text{Engine},2}$	-0.50	0.00	0.00	-0.25	0.00	0.00
66	Approx. Engine 2 Length (mm)	3,937.00	0.00	0.00	5,190.00	0.00	0.00
67	Approx. Engine 2 Diameter (mm)	990.00	0.00	0.00	2,190.00	0.00	0.00

Table 57 Some of the NASA SUGAR Aircraft design parameters used in the design tool to generate the baseline SBW aircraft for the parametric study.

No	Parameter	Value	Units	No	Parameter	Value	Units
1	Aircraft Name	NASA SUGAR		22	Wing total mass	7,561.38	kg
2	Root chord	3,309.94	mm	23	Wing box actual mass	3,615.00	kg
3	Tip chord	1,146.40	mm	24	Total engine count	2.00	
4	ND front spar location at the wing root as a fraction of wing section chord.	0.15		25	Engine mounting type. 1 for wing-mounted, 2 for fuselage mounted.	1.00	
5	ND front spar location at the wing tip as a fraction of wing section chord.	0.25		26	Engine total mass	5,996.5	kg
6	ND rear spar location at the wing root as a fraction of wing section chord.	0.60		27	ND spanwise Engine-1 position as a fraction of half wingspan	0.3	
7	ND rear spar location at the wing tip as a fraction of wing section chord.	0.60		28	ND chordwise Engine-1 position as a fraction of local wing section chord	-0.25	
8	Wing thickness to chord ratio at the root.	0.15		29	ND, z-axis, Engine-1 position as a fraction of local wing section thickness	-3.00	
9	Wing thickness to chord ratio at the tip.	0.15		30	Fuselage length	42,587.67	mm
10	Wing sweep angle	13.47		31	Fuselage diameter	3,776.98	mm
11	Wing dihedral	-1.50		32	Landing gear mounting type. 1 for wing-mounted, 2 for fuselage mounted.	2.00	
12	Stringer count	3.00		33	Main LG Weight	1,163.46	kg
13	Wingspan	51,798.25	mm	34	Wing maximum lift coefficient	1.50	
14	TOGW	68,038.86	kg	35	Service Ceiling	12,435.84	m
15	OEW	39,598.61	kg	36	Aspect Ratio Actual	19.54	
16	Design diving speed	165.00	m/s				
17	Design diving Mach	0.76					
18	Design cruise speed	150.00	m/s				
19	Design cruise Mach	0.71					
20	Wing-stored fuel mass	12,485.63	kg				
21	Total fuel volume	20.50	m ³				

B.2 2D Views of the Aircraft Used for Validation

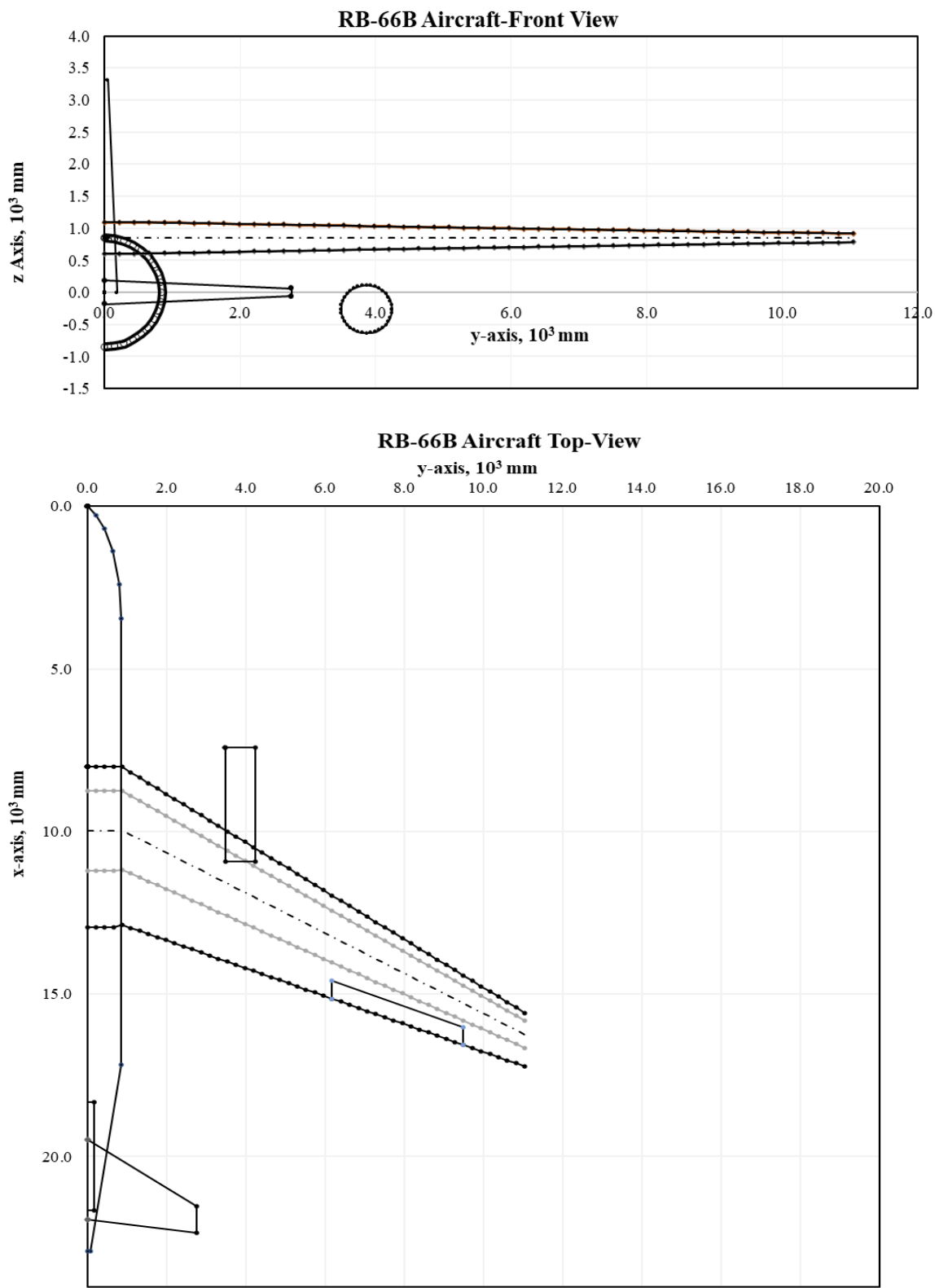


Figure 108 The top and the front views of RB-66B Aircraft used for validation.

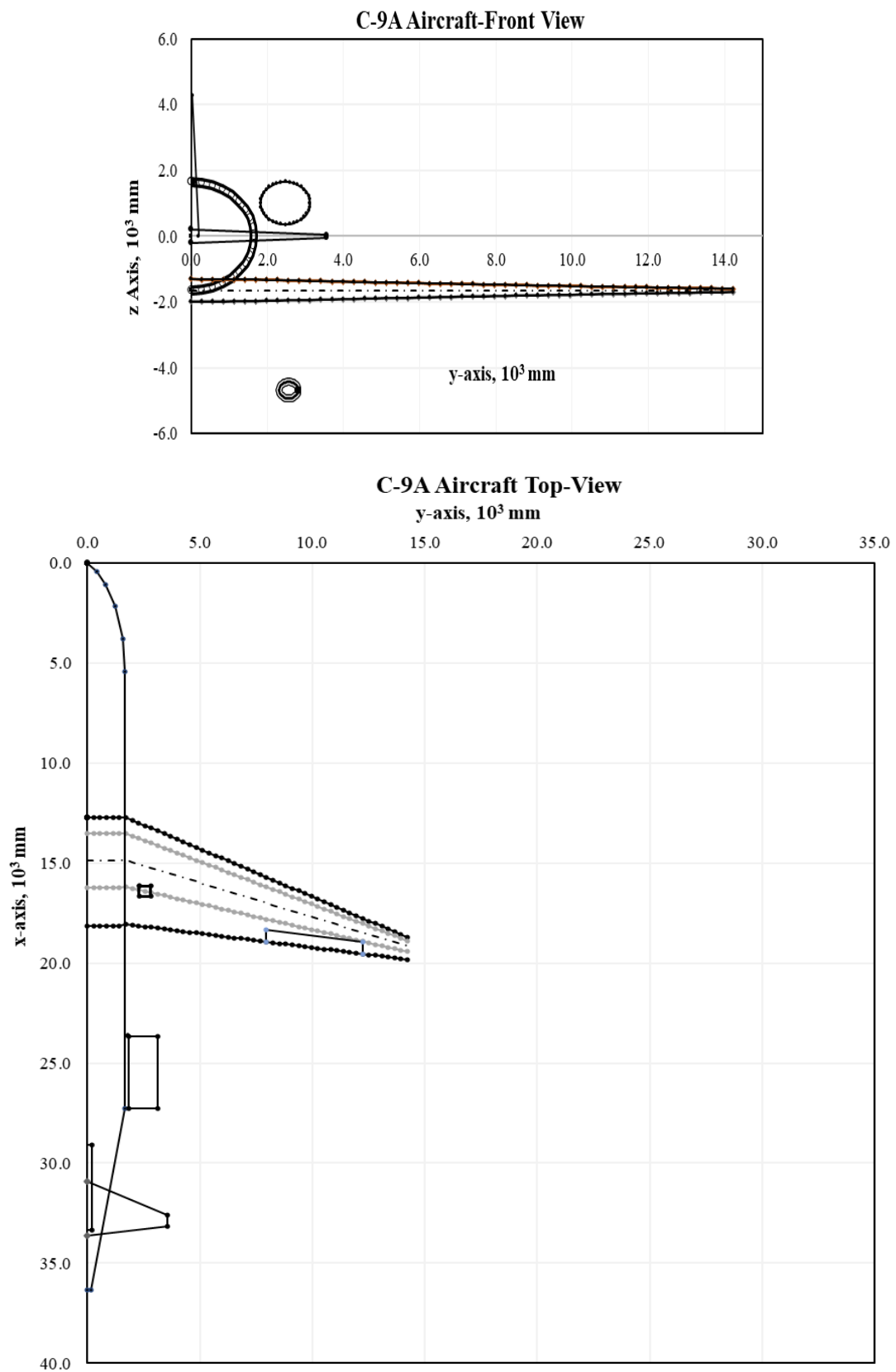


Figure 109 The top and the front views of C-9A Aircraft used for validation.

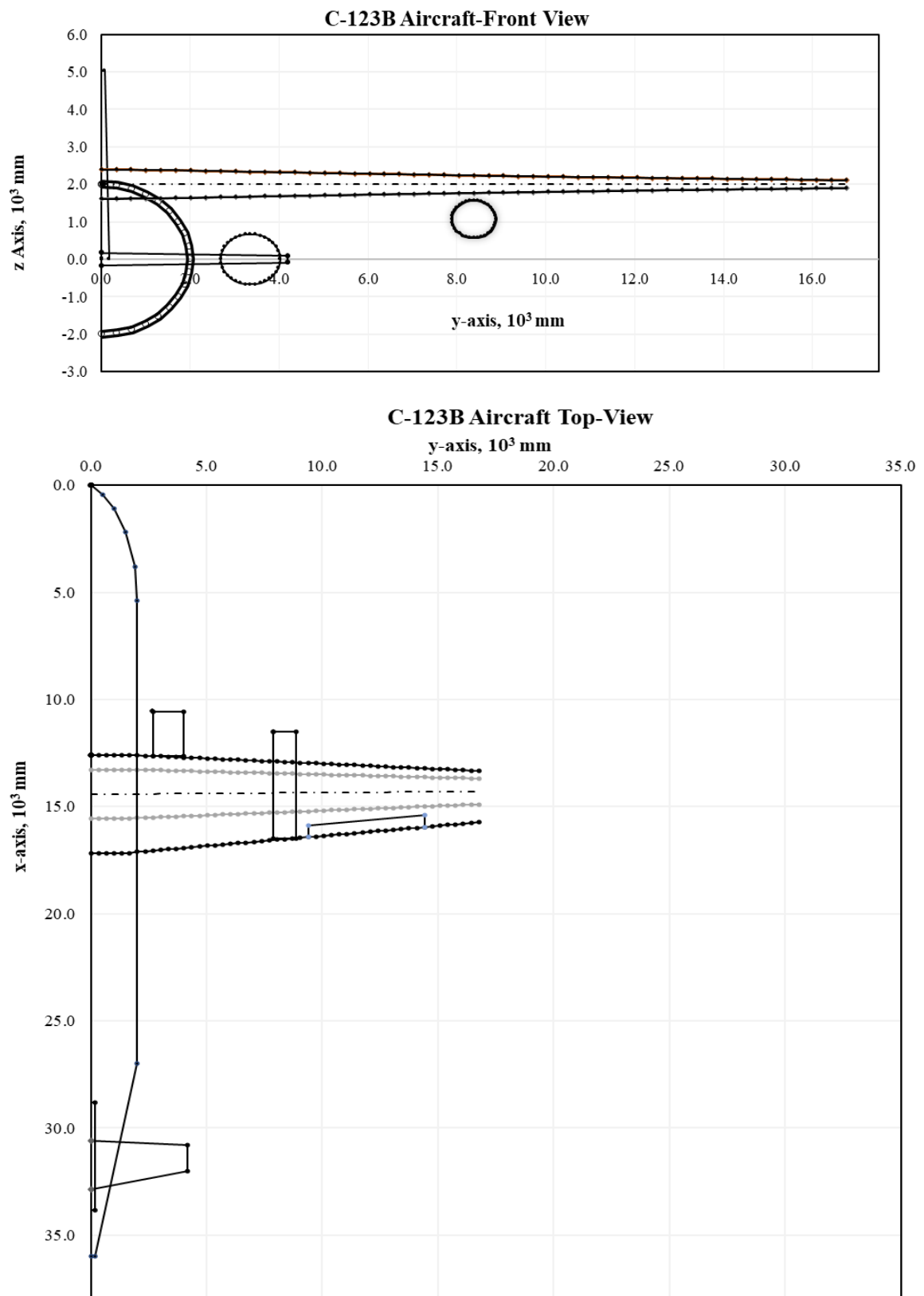


Figure 110 The top and the front views of C-123B Aircraft used for validation.

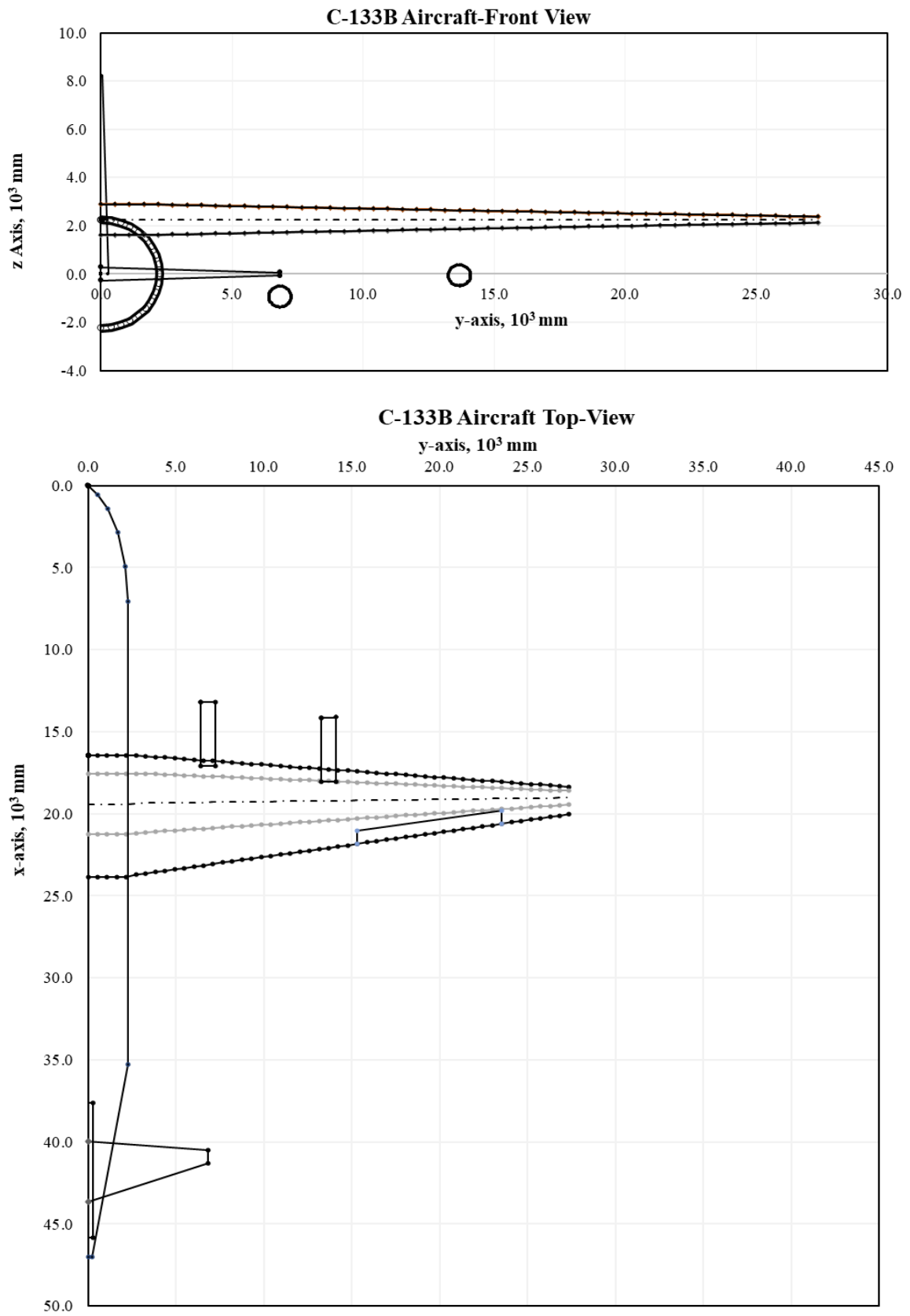


Figure 111 The top and the front views of C-133B Aircraft used for validation.

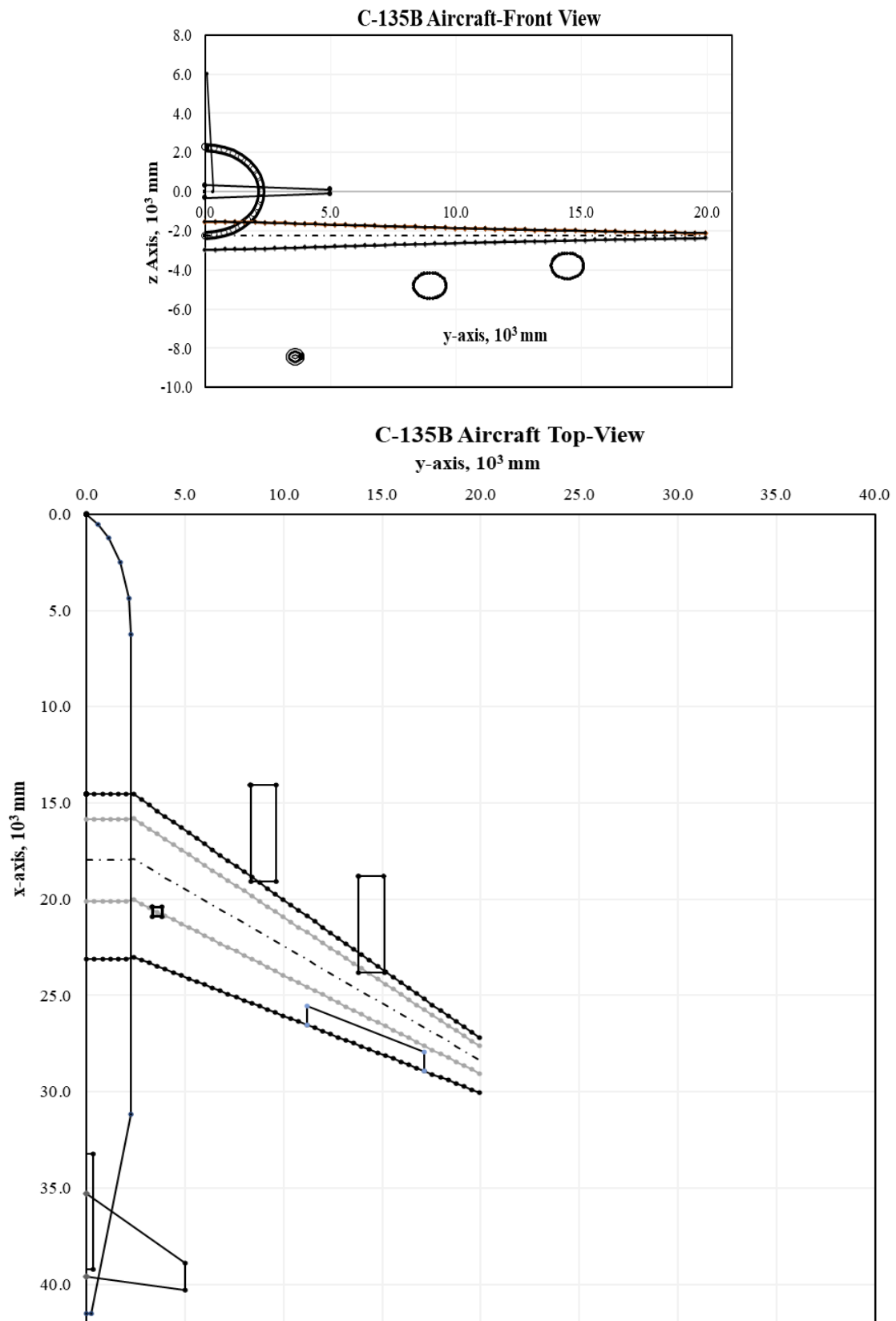


Figure 112 The top and the front views of C-135B Aircraft used for validation.

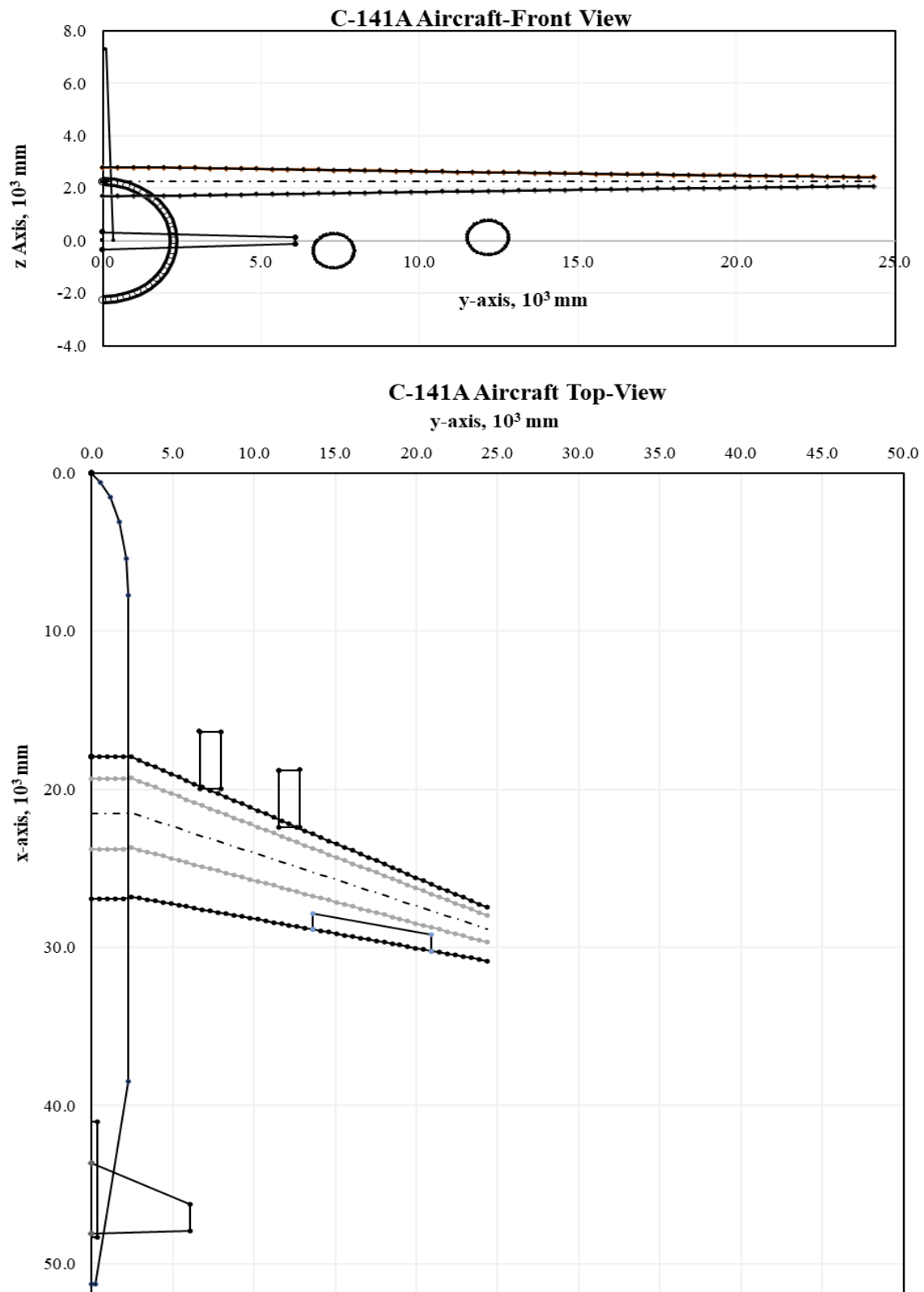


Figure 113 The top and the front views of C-141A Aircraft used for validation.

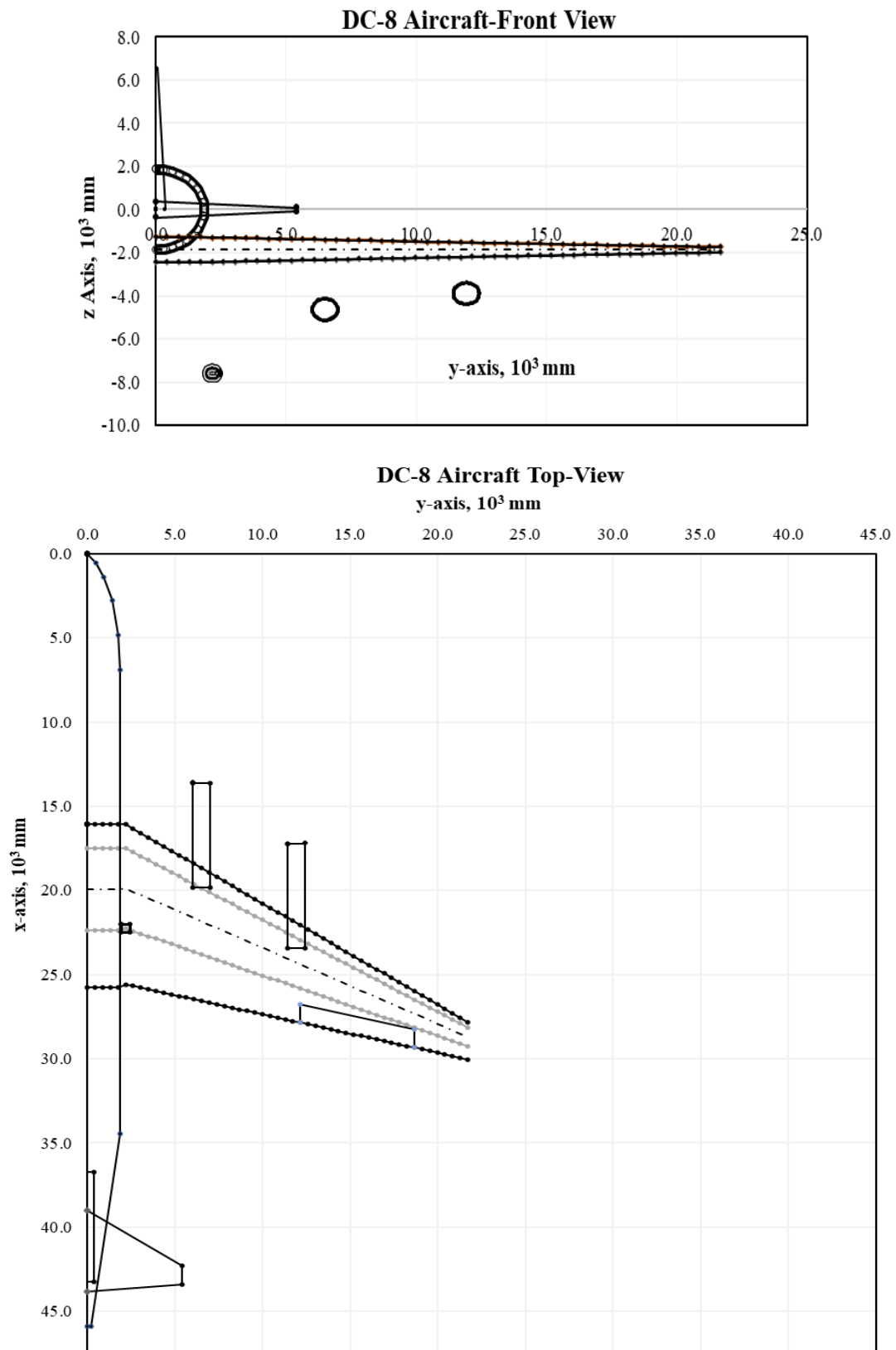


Figure 114 The top and the front views of DC-8 Aircraft used for validation.

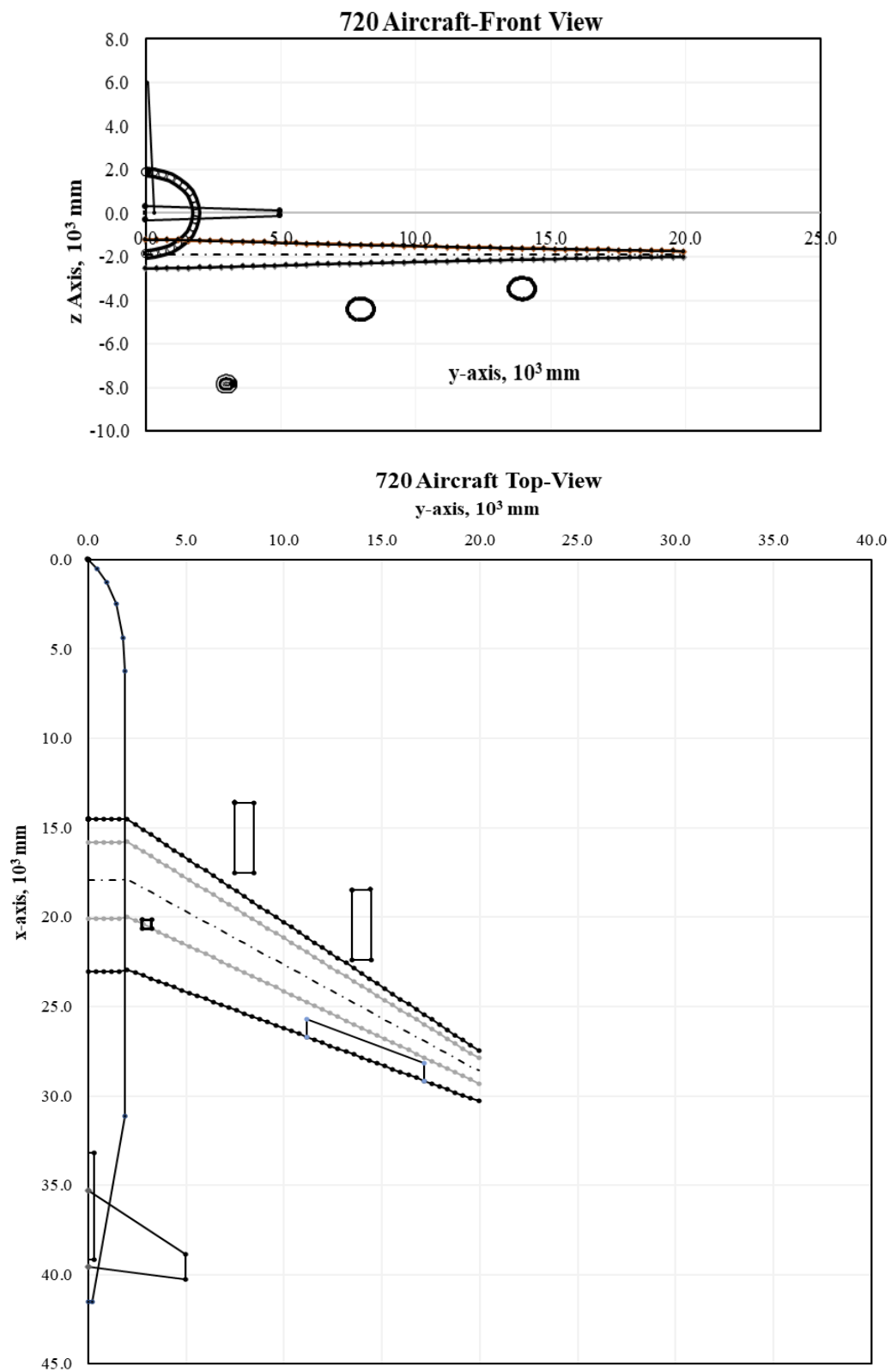


Figure 115 The top and the front views of 720 Aircraft used for validation.

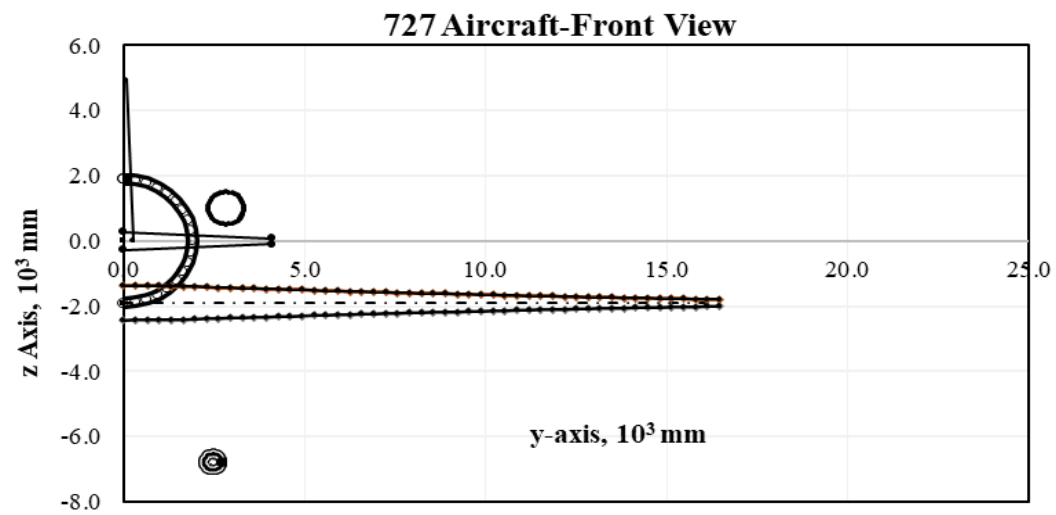
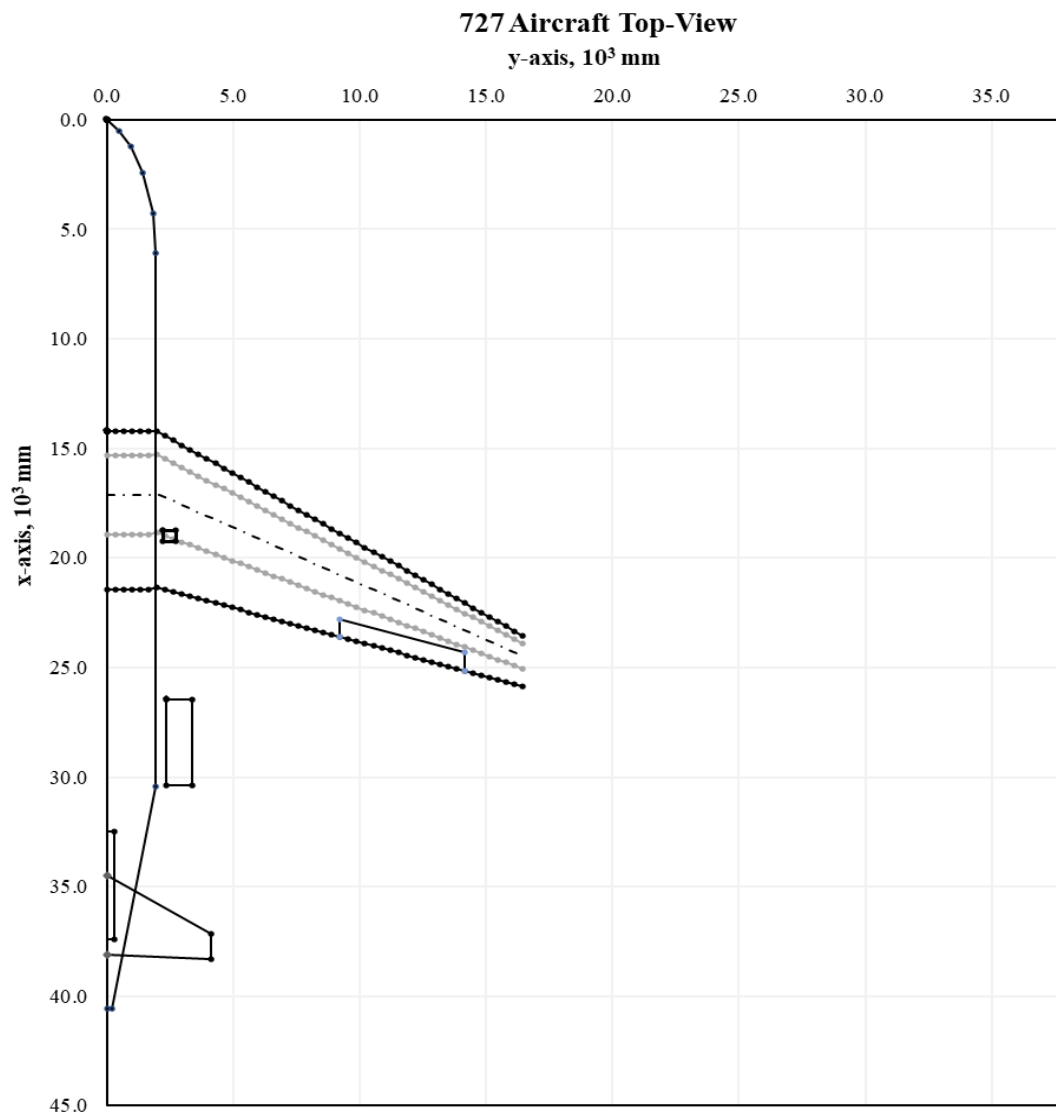


Figure 116 The top and the front views of 727 Aircraft used for validation.

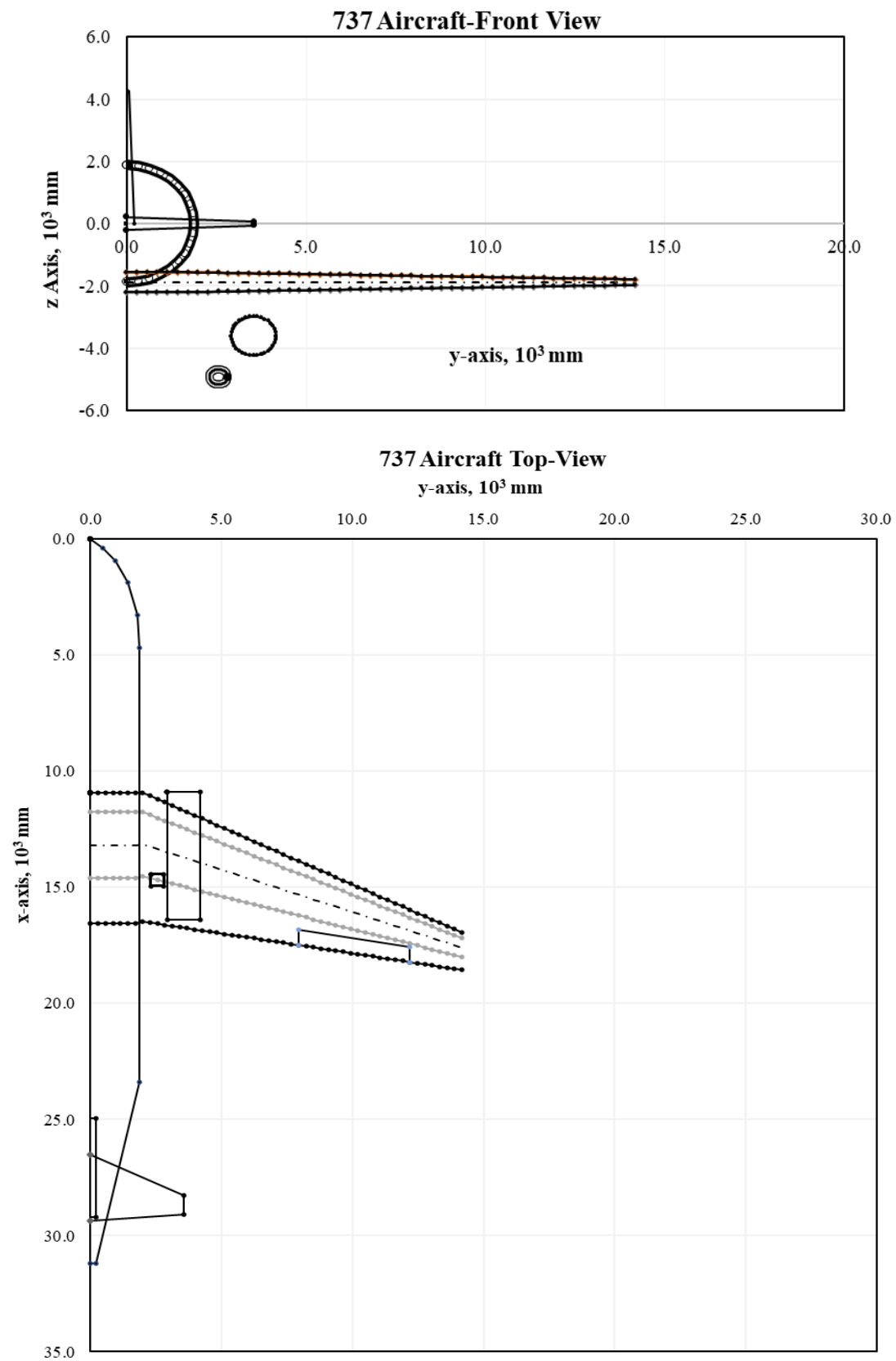


Figure 117 The top and the front views of 737 Aircraft used for validation.

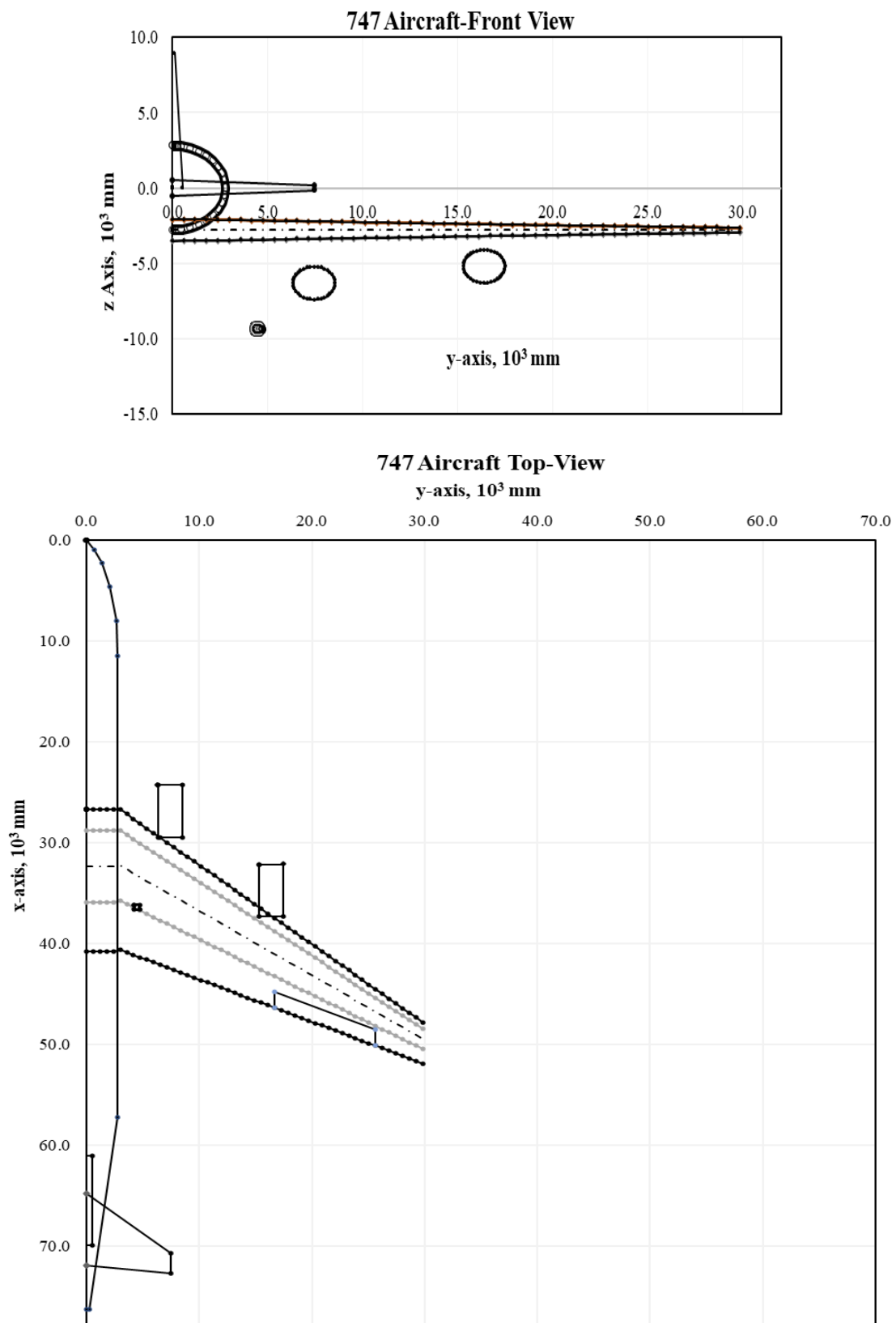


Figure 118 The top and the front views of 747 Aircraft used for validation.

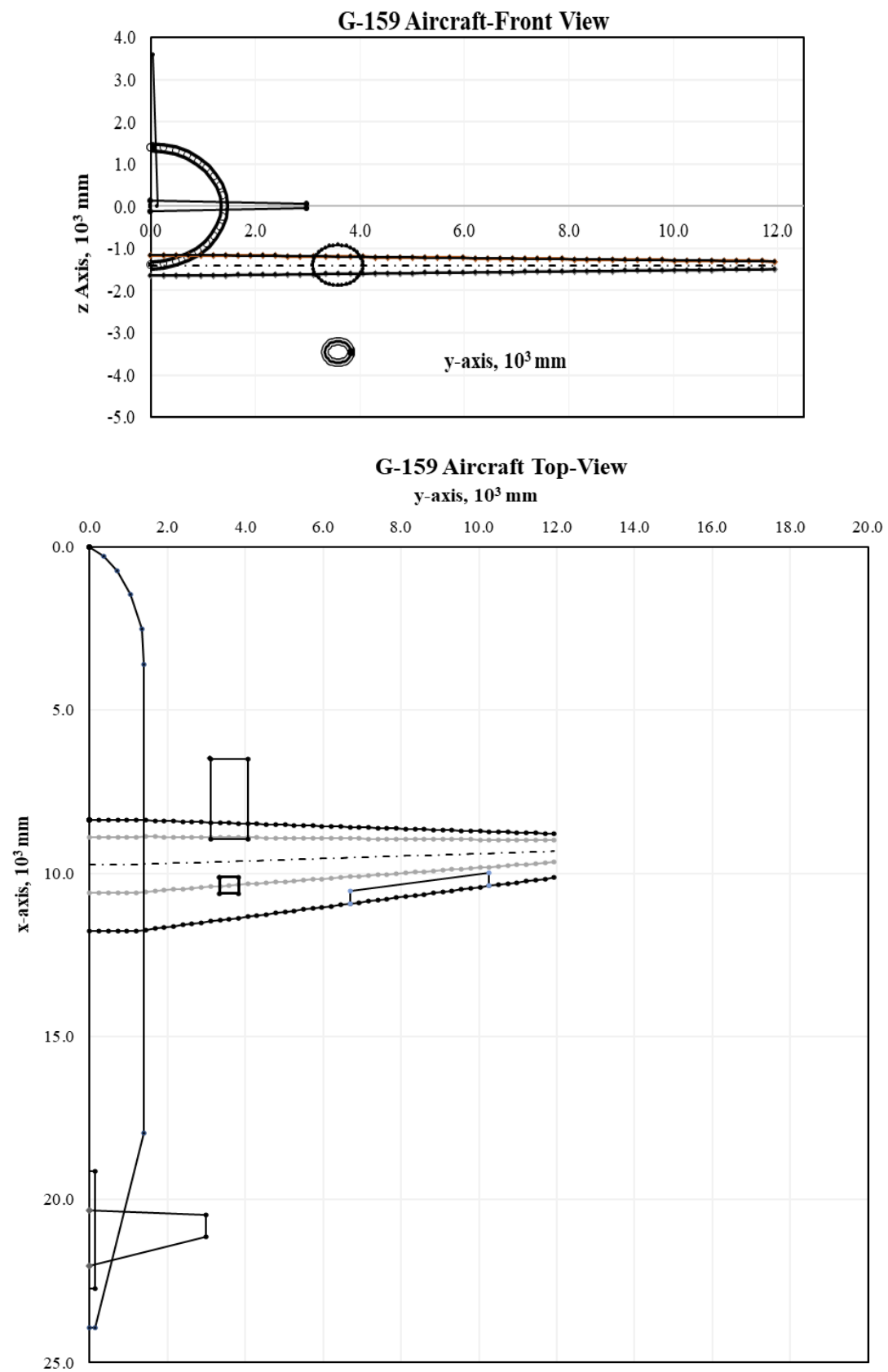


Figure 119 The top and the front views of G-159 Aircraft used for validation.

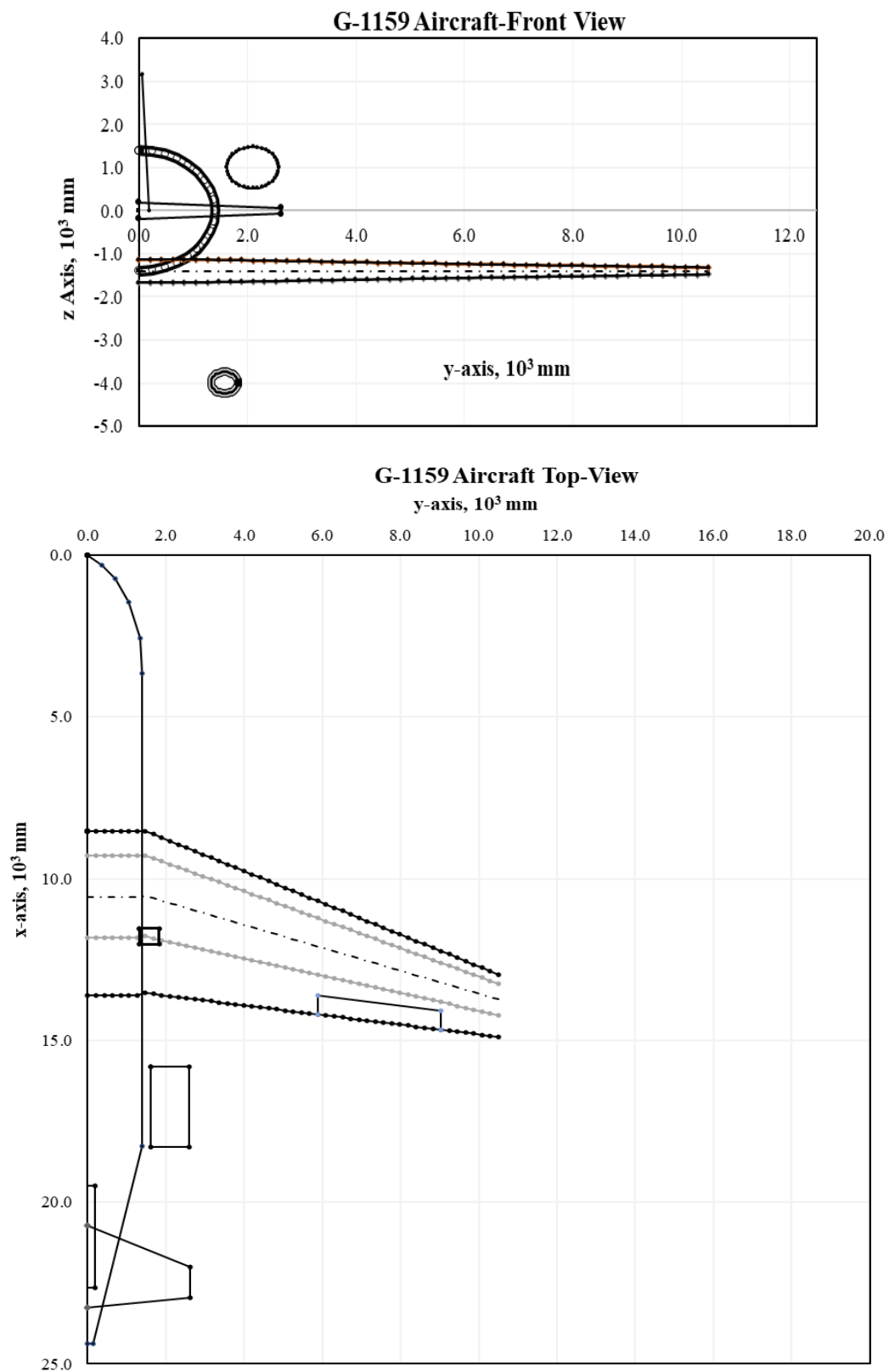


Figure 120 The top and the front views of G-1159 Aircraft used for validation.

B.3 Wing Box Mass Breakdown of Some Baseline Aircraft

Table 58 Estimated wing box mass breakdown of NASA SUGAR Aircraft and the change of the masses with the variation of stringer counts.

Stringer Counts	Shear Resistant Material, (kg)	Bending Resistant Material, (kg)	Rib Mass (Torenbeek), (kg)	Aeroelasticity, (kg)	Total WB Structure Mass, (kg)
4	1,035.9	2,131.8	125.0	456.7	3,749.4
5	969.6	2,307.2	125.0	456.7	3,858.5
6	908.6	2,427.9	125.0	456.7	3,918.2
7	863.2	2,549.7	125.0	456.7	3,994.5
8	825.0	2,626.3	125.0	456.7	4,033.0
9	810.0	2,710.8	125.0	456.7	4,102.5
10	794.7	2,757.6	125.0	456.7	4,134.0
11	783.4	2,834.5	125.0	456.7	4,199.6
12	775.0	2,876.5	125.0	456.7	4,233.2
13	780.3	2,925.9	125.0	456.7	4,287.9
14	776.0	2,960.6	125.0	456.7	4,318.2
15	771.2	3,002.7	125.0	456.7	4,355.7
16	765.8	3,030.6	125.0	456.7	4,378.2
17	766.4	3,074.8	125.0	456.7	4,423.0
18	771.2	3,082.4	125.0	456.7	4,435.4
19	768.2	3,125.5	125.0	456.7	4,475.4
20	764.6	3,134.3	125.0	456.7	4,480.7
21	764.8	3,163.6	125.0	456.7	4,510.2
22	769.8	3,173.1	125.0	456.7	4,524.5
23	767.0	3,212.1	125.0	456.7	4,560.9
24	764.7	3,215.0	125.0	456.7	4,561.4
25	761.3	3,242.4	125.0	456.7	4,585.4
26	765.2	3,250.5	125.0	456.7	4,597.4
27	767.5	3,266.1	125.0	456.7	4,615.3
28	765.0	3,267.0	125.0	456.7	4,613.7
29	762.6	3,304.1	125.0	456.7	4,648.4
30	763.6	3,297.1	125.0	456.7	4,642.5

Table 59 Estimated wing box mass breakdown of SUGAR-CANT Aircraft and the change of the masses with the variation of stringer counts.

Stringer Counts	Shear Resistant Material, (kg)	Bending Resistant Material, (kg)	Rib Mass (Torenbeek), (kg)	Aeroelasticity, (kg)	Total WB Structure Mass, (kg)
4.0	1,119.3	3,118.4	125.0	456.7	4,819.3
5.0	1,068.7	3,497.7	125.0	456.7	5,148.1
6.0	1,022.6	3,480.4	125.0	456.7	5,084.7
7.0	984.2	3,756.2	125.0	456.7	5,322.1
8.0	968.0	3,710.2	125.0	456.7	5,259.9
9.0	945.2	3,927.2	125.0	456.7	5,454.1
10.0	930.5	3,861.8	125.0	456.7	5,374.0
11.0	918.9	4,034.5	125.0	456.7	5,535.1
12.0	911.5	3,979.2	125.0	456.7	5,472.4
13.0	906.5	4,120.7	125.0	456.7	5,608.8
14.0	902.6	4,062.9	125.0	456.7	5,547.3
15.0	897.2	4,189.4	125.0	456.7	5,668.3
16.0	901.9	4,126.7	125.0	456.7	5,610.4
17.0	904.3	4,236.9	125.0	456.7	5,722.9
18.0	899.9	4,185.2	125.0	456.7	5,666.9
19.0	895.6	4,277.7	125.0	456.7	5,755.1
20.0	890.9	4,225.2	125.0	456.7	5,697.8
21.0	886.1	4,316.5	125.0	456.7	5,784.4
22.0	882.3	4,268.3	125.0	456.7	5,732.4
23.0	877.8	4,339.4	125.0	456.7	5,798.9
24.0	873.8	4,296.4	125.0	456.7	5,752.0
25.0	877.2	4,371.1	125.0	456.7	5,830.0
26.0	880.5	4,322.1	125.0	456.7	5,784.3
27.0	874.8	4,389.9	125.0	456.7	5,846.3
28.0	871.4	4,352.7	125.0	456.7	5,805.8
29.0	867.0	4,404.7	125.0	456.7	5,853.4
30.0	862.8	4,367.0	125.0	456.7	5,811.5

Table 60 Estimated wing box mass breakdown of Boeing 747 Aircraft and the change of the masses with the variation of stringer counts.

Stringer Counts	Shear Resistant Material, (kg)	Bending Resistant Material, (kg)	Rib Mass (Torenbeek), (kg)	Aeroelasticity, (kg)	Total WB Structure Mass, (kg)
4.0	20,484.5	5,301.4	1,483.5	371.5	27,640.9
5.0	20,128.0	5,964.8	1,483.5	371.5	27,947.9
6.0	19,684.6	5,946.9	1,483.5	371.5	27,486.5
7.0	19,237.9	6,416.6	1,483.5	371.5	27,509.6
8.0	18,796.8	6,329.4	1,483.5	371.5	26,981.2
9.0	18,328.7	6,668.7	1,483.5	371.5	26,852.5
10.0	17,872.1	6,582.3	1,483.5	371.5	26,309.4
11.0	17,436.5	6,870.7	1,483.5	371.5	26,162.2
12.0	17,008.5	6,757.4	1,483.5	371.5	25,620.9
13.0	16,563.7	7,016.9	1,483.5	371.5	25,435.6
14.0	16,169.0	6,912.1	1,483.5	371.5	24,936.1
15.0	15,783.5	7,136.2	1,483.5	371.5	24,774.7
16.0	15,451.8	7,023.7	1,483.5	371.5	24,330.5
17.0	15,095.9	7,209.9	1,483.5	371.5	24,160.8
18.0	14,771.6	7,111.2	1,483.5	371.5	23,737.8
19.0	14,466.1	7,271.1	1,483.5	371.5	23,592.2
20.0	14,163.4	7,187.5	1,483.5	371.5	23,205.9
21.0	13,892.7	7,327.0	1,483.5	371.5	23,074.7
22.0	13,637.4	7,245.2	1,483.5	371.5	22,737.6
23.0	13,379.7	7,379.7	1,483.5	371.5	22,614.4
24.0	13,137.5	7,300.7	1,483.5	371.5	22,293.2
25.0	12,909.8	7,403.7	1,483.5	371.5	22,168.5
26.0	12,695.9	7,339.0	1,483.5	371.5	21,889.9
27.0	12,488.8	7,456.2	1,483.5	371.5	21,800.0
28.0	12,298.9	7,377.2	1,483.5	371.5	21,531.2
29.0	12,106.5	7,485.2	1,483.5	371.5	21,446.7
30.0	11,929.3	7,417.9	1,483.5	371.5	21,202.2

Table 61 Estimated wing box mass breakdown of Boeing 737 Aircraft and the change of the masses with the variation of stringer counts.

Stringer Counts	Shear Resistant Material, (kg)	Bending Resistant Material, (kg)	Rib Mass (Torenbeek), (kg)	Aeroelasticity, (kg)	Total WB Structure Mass, (kg)
4.0	1,733.9	449.1	216.4	69.1	2,468.5
5.0	1,691.4	504.1	216.4	69.1	2,481.1
6.0	1,644.2	503.3	216.4	69.1	2,432.9
7.0	1,595.3	541.4	216.4	69.1	2,422.2
8.0	1,546.0	535.2	216.4	69.1	2,366.8
9.0	1,496.1	564.9	216.4	69.1	2,346.6
10.0	1,449.4	557.6	216.4	69.1	2,292.6
11.0	1,405.5	581.2	216.4	69.1	2,272.2
12.0	1,364.3	572.4	216.4	69.1	2,222.2
13.0	1,324.3	592.5	216.4	69.1	2,202.3
14.0	1,286.4	584.8	216.4	69.1	2,156.7
15.0	1,252.1	601.8	216.4	69.1	2,139.5
16.0	1,218.7	593.7	216.4	69.1	2,097.9
17.0	1,187.8	608.5	216.4	69.1	2,081.9
18.0	1,160.1	600.6	216.4	69.1	2,046.2
19.0	1,134.1	615.0	216.4	69.1	2,034.6
20.0	1,111.7	606.8	216.4	69.1	2,004.0
21.0	1,088.3	619.3	216.4	69.1	1,993.1
22.0	1,066.1	612.6	216.4	69.1	1,964.2
23.0	1,045.7	623.7	216.4	69.1	1,955.0
24.0	1,026.3	615.8	216.4	69.1	1,927.6
25.0	1,007.4	626.6	216.4	69.1	1,919.5
26.0	990.2	620.4	216.4	69.1	1,896.2
27.0	973.7	629.6	216.4	69.1	1,888.7
28.0	957.9	623.4	216.4	69.1	1,866.8
29.0	942.5	632.7	216.4	69.1	1,860.6
30.0	927.9	625.1	216.4	69.1	1,838.5

Appendix C Critical Loading Cases

Table 62 Twenty-one critical loading cases, which sized a minimum of 1000 wing components out of 197,367 in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Positive	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	7	2,253	Manoeuvring	No Rolling	0	0	Positive	10,668	0	100	V_C
4	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
5	17	1,740	Gust	No Rolling	0	0	Positive	0	100	100	V_C
6	32	3,151	Gust	No Rolling	0	0	Positive	0	0	0	V_C
7	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
8	41	3,618	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C
9	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
10	53	1,796	Gust	No Rolling	0	0	Positive	6,096	100	100	V_C
11	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
12	104	1,229	Gust	No Rolling	0	0	Positive	12,192	0	0	V_C
13	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
14	343	3,835	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	0	V_C
15	360	3,939	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_C
16	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
17	444	1,937	Manoeuvring	Steady	Rolling Down	n0	Positive	0	0	100	V_C
18	445	1,201	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	100	V_C
19	451	1,427	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	0	V_C
20	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C
21	457	1,985	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	100	V_C

Table 63 Twenty-seven critical loading cases, which sized a minimum of 500 wing components out of 197,367 in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Positive	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	7	2,253	Manoeuvring	No Rolling	0	0	Positive	10,668	0	100	V_C
4	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
5	17	1,740	Gust	No Rolling	0	0	Positive	0	100	100	V_C
6	32	3,151	Gust	No Rolling	0	0	Positive	0	0	0	V_C
7	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
8	41	3,618	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C
9	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
10	53	1,796	Gust	No Rolling	0	0	Positive	6,096	100	100	V_C
11	59	989	Gust	No Rolling	0	0	Positive	6,096	100	0	V_C
12	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
13	86	1,000	Gust	No Rolling	0	0	Positive	9,144	0	0	V_C
14	104	1,229	Gust	No Rolling	0	0	Positive	12,192	0	0	V_C
15	140	967	Gust	No Rolling	0	0	Negative	3,048	0	0	V_C
16	253	783	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	0	100	V_C
17	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
18	343	3,835	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	0	V_C
19	348	561	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	0	100	V_C
20	360	3,939	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_C
21	362	678	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_D
22	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
23	444	1,937	Manoeuvring	Steady	Rolling Down	n0	Positive	0	0	100	V_C
24	445	1,201	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	100	V_C
25	451	1,427	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	0	V_C
26	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C
27	457	1,985	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	100	V_C

Table 64 Fifty-five critical loading cases, which sized a minimum of 100 wing components out of 197,367 in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Positive	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	6	257	Manoeuvring	No Rolling	0	0	Positive	10,668	100	0	V_C
4	7	2,253	Manoeuvring	No Rolling	0	0	Positive	10,668	0	100	V_C
5	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
6	17	1,740	Gust	No Rolling	0	0	Positive	0	100	100	V_C
7	18	111	Gust	No Rolling	0	0	Positive	0	100	100	V_D
8	23	175	Gust	No Rolling	0	0	Positive	0	100	0	V_C
9	26	111	Gust	No Rolling	0	0	Positive	0	0	100	V_C
10	32	3,151	Gust	No Rolling	0	0	Positive	0	0	0	V_C
11	34	500	Gust	No Rolling	0	0	Positive	3,048	100	100	V_B
12	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
13	41	3,618	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C
14	44	485	Gust	No Rolling	0	0	Positive	3,048	0	100	V_C
15	49	442	Gust	No Rolling	0	0	Positive	3,048	0	0	V_B
16	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
17	52	272	Gust	No Rolling	0	0	Positive	6,096	100	100	V_B
18	53	1,796	Gust	No Rolling	0	0	Positive	6,096	100	100	V_C
19	59	989	Gust	No Rolling	0	0	Positive	6,096	100	0	V_C
20	62	217	Gust	No Rolling	0	0	Positive	6,096	0	100	V_C
21	67	200	Gust	No Rolling	0	0	Positive	6,096	0	0	V_B
22	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
23	77	165	Gust	No Rolling	0	0	Positive	9,144	100	0	V_C
24	86	1,000	Gust	No Rolling	0	0	Positive	9,144	0	0	V_C
25	95	185	Gust	No Rolling	0	0	Positive	12,192	100	0	V_C
26	104	1,229	Gust	No Rolling	0	0	Positive	12,192	0	0	V_C
27	122	281	Gust	No Rolling	0	0	Negative	0	0	0	V_C
28	125	201	Gust	No Rolling	0	0	Negative	3,048	100	100	V_C
29	131	115	Gust	No Rolling	0	0	Negative	3,048	100	0	V_C
30	140	967	Gust	No Rolling	0	0	Negative	3,048	0	0	V_C
31	158	221	Gust	No Rolling	0	0	Negative	6,096	0	0	V_C
32	247	204	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	0	0	V_C

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
33	253	783	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	0	100	V_C
34	264	246	Manoeuvring	Initiation	Rolling Up	n0	Positive	0	100	100	V_C
35	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
36	343	3,835	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	0	V_C
37	348	561	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	0	100	V_C
38	349	109	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	100	V_C
39	354	188	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	0	V_C
40	355	116	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	100	0	V_C
41	360	3,939	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_C
42	361	121	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	100	100	V_C
43	362	678	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_D
44	384	230	Manoeuvring	Arrest	Rolling Down	n0	Positive	12,192	100	100	V_C
45	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
46	441	158	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_D
47	444	1,937	Manoeuvring	Steady	Rolling Down	n0	Positive	0	0	100	V_C
48	445	1,201	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	100	V_C
49	450	273	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	0	V_C
50	451	1,427	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	0	V_C
51	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C
52	457	1,985	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	100	V_C
53	458	149	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_D
54	463	367	Manoeuvring	Steady	Rolling Down	n2\3	Positive	12,192	0	0	V_C
55	480	353	Manoeuvring	Steady	Rolling Down	n0	Positive	12,192	100	100	V_C

Table 65 Sixty-three critical loading cases, sized at least 50 wing components out of 197,367 in 1941 aircraft samples.

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
1	3	2,612	Taxi	No Rolling	0	0	Positive	0	100	100	V_B
2	4	52,461	Manoeuvring	No Rolling	0	0	Positive	10,668	100	100	V_C
3	6	257	Manoeuvring	No Rolling	0	0	Positive	10,668	100	0	V_C
4	7	2,253	Manoeuvring	No Rolling	0	0	Positive	10,668	0	100	V_C
5	9	6,717	Manoeuvring	No Rolling	0	0	Positive	10,668	0	0	V_C
6	17	1,740	Gust	No Rolling	0	0	Positive	0	100	100	V_C
7	18	111	Gust	No Rolling	0	0	Positive	0	100	100	V_D
8	23	175	Gust	No Rolling	0	0	Positive	0	100	0	V_C
9	26	111	Gust	No Rolling	0	0	Positive	0	0	100	V_C
10	32	3,151	Gust	No Rolling	0	0	Positive	0	0	0	V_C
11	33	60	Gust	No Rolling	0	0	Positive	0	0	0	V_D
12	34	500	Gust	No Rolling	0	0	Positive	3,048	100	100	V_B
13	35	7,711	Gust	No Rolling	0	0	Positive	3,048	100	100	V_C
14	41	3,618	Gust	No Rolling	0	0	Positive	3,048	100	0	V_C
15	44	485	Gust	No Rolling	0	0	Positive	3,048	0	100	V_C
16	49	442	Gust	No Rolling	0	0	Positive	3,048	0	0	V_B
17	50	23,575	Gust	No Rolling	0	0	Positive	3,048	0	0	V_C
18	52	272	Gust	No Rolling	0	0	Positive	6,096	100	100	V_B
19	53	1,796	Gust	No Rolling	0	0	Positive	6,096	100	100	V_C
20	59	989	Gust	No Rolling	0	0	Positive	6,096	100	0	V_C
21	62	217	Gust	No Rolling	0	0	Positive	6,096	0	100	V_C
22	67	200	Gust	No Rolling	0	0	Positive	6,096	0	0	V_B
23	68	6,760	Gust	No Rolling	0	0	Positive	6,096	0	0	V_C
24	77	165	Gust	No Rolling	0	0	Positive	9,144	100	0	V_C
25	86	1,000	Gust	No Rolling	0	0	Positive	9,144	0	0	V_C
26	95	185	Gust	No Rolling	0	0	Positive	12,192	100	0	V_C
27	104	1,229	Gust	No Rolling	0	0	Positive	12,192	0	0	V_C
28	122	281	Gust	No Rolling	0	0	Negative	0	0	0	V_C
29	125	201	Gust	No Rolling	0	0	Negative	3,048	100	100	V_C
30	131	115	Gust	No Rolling	0	0	Negative	3,048	100	0	V_C
31	140	967	Gust	No Rolling	0	0	Negative	3,048	0	0	V_C
32	143	72	Gust	No Rolling	0	0	Negative	6,096	100	100	V_C

No	Case Number	Number of Sized Components	Loading Case Type	Rolling Case Type	Rolling Direction	Manoeuvring in Rolling Case	Manoeuvring Direction	Altitude (m)	Payload (%)	Fuel (%)	Speed
33	149	58	Gust	No Rolling	0	0	Negative	6,096	100	0	V_C
34	158	221	Gust	No Rolling	0	0	Negative	6,096	0	0	V_C
35	176	64	Gust	No Rolling	0	0	Negative	9,144	0	0	V_C
36	194	66	Gust	No Rolling	0	0	Negative	12,192	0	0	V_C
37	247	204	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	0	0	V_C
38	253	783	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	0	100	V_C
39	264	246	Manoeuvring	Initiation	Rolling Up	n0	Positive	0	100	100	V_C
40	265	14,094	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_C
41	267	84	Manoeuvring	Initiation	Rolling Up	n2\3	Positive	0	100	100	V_D
42	343	3,835	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	0	V_C
43	348	561	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	0	100	V_C
44	349	109	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	0	100	V_C
45	350	58	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	0	100	V_D
46	354	188	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	0	V_C
47	355	116	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	100	0	V_C
48	360	3,939	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_C
49	361	121	Manoeuvring	Arrest	Rolling Down	n2\3	Positive	0	100	100	V_C
50	362	678	Manoeuvring	Arrest	Rolling Down	n0	Positive	0	100	100	V_D
51	384	230	Manoeuvring	Arrest	Rolling Down	n0	Positive	12,192	100	100	V_C
52	439	25,200	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_C
53	441	158	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	0	V_D
54	444	1,937	Manoeuvring	Steady	Rolling Down	n0	Positive	0	0	100	V_C
55	445	1,201	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	0	100	V_C
56	450	273	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	0	V_C
57	451	1,427	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	0	V_C
58	456	16,858	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_C
59	457	1,985	Manoeuvring	Steady	Rolling Down	n2\3	Positive	0	100	100	V_C
60	458	149	Manoeuvring	Steady	Rolling Down	n0	Positive	0	100	100	V_D
61	463	367	Manoeuvring	Steady	Rolling Down	n2\3	Positive	12,192	0	0	V_C
62	469	71	Manoeuvring	Steady	Rolling Down	n2\3	Positive	12,192	0	100	V_C
63	480	353	Manoeuvring	Steady	Rolling Down	n0	Positive	12,192	100	100	V_C

