

Conceptual Design and Optimization Methodology for Box Wing Aircraft

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Unto the king eternal, immortal, invisible the only wise God be honour and glory
forever and ever, amen. I Tim 1.17

Abstract

A conceptual design optimization methodology was developed for a medium range box wing aircraft. A baseline conventional cantilever wing aircraft designed for the same mission and payload was also optimized alongside a baseline box wing aircraft. An empirical formula for the mass estimation of the fore and aft wings of the box wing aircraft was derived by relating conventional cantilever wings to box wing aircraft wings. The results indicate that the fore and aft wings would use the same correction coefficient and that the aft wing would be lighter than the fore wing on the medium range box wing aircraft because of reduced sweep.

As part of the methodology, a computational study was performed to analyze different wing/tip fin fixities using a statically loaded idealized box wing configuration. The analyses determined the best joint fixity by comparing the stress distributions in finite element torsion box models in addition to aerodynamic requirements. The analyses indicates that the rigid joint is the most suitable.

Studies were also performed to investigate the structural implications of changing only the tip fin inclinations on the box wing aircraft. Tip fin inclination refers to the angle the tip fin makes to the vertical body axis of the aircraft. No significant variations in wing structural design drivers as a function of tip fin inclination were observed.

Stochastic and deterministic optimization routines were performed on the baseline box wing aircraft using the methodology developed where the variables were wing area, average thickness to chord ratio and sweep angle. The conventional aircraft design showed similar performance and characteristics to the equivalent in-service aircraft thereby providing some validation to the methodology and the results for the box wing aircraft.

Longitudinal stability investigations showed that the extra fuel capacity of the box wing in the fins could be used to reduce trim drag. The short period oscillation of the conventional cantilever wing aircraft was found to be satisfactory but the box wing aircraft was found to be unacceptable hence requiring stability augmentation systems. The field and flight performance of the box wing showed to be better than the conventional cantilever wing aircraft. Overall, the economic advantages of the box wing aircraft over the conventional cantilever wing aircraft improve with increase in fuel price making the box wing a worthy replacement for the conventional cantilever wing aircraft.

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Nomenclature

a	Wing body lift curve slope
a_o	Aft wing zero incidence lift coefficient
a_1	Aft wing lift curve slope
a_2	Elevator lift curve slope
a_3	Elevator tab lift curve slope
AFC	Aircraft fuel capacity
AoA	Angle of Attack
APU	Auxiliary Power Unit
AR	Wing aspect ratio
$AWTM$	Aft wing torsion box mass
AW	Aft wing
b	Wing span
BF	Block fuel
c_A	Aft wing mean aerodynamic chord
CAD	Computer aided design
CAE	Computer aided engineering
CAM	Computer aided manufacture
C_D	Coefficient of drag
C_{D_0}	Zero lift drag parameter
c_F	Fore wing mean aerodynamic chord
CFD	Computational Fluid Dynamics
cg	Center of gravity
C_l	Lift coefficient
C_{lA}	Aft wing lift coefficient
C_{lF}	Fore wing lift coefficient
C_m	Pitching moment coefficient
C_{mA}	Aft wing pitching moment coefficient
C_{mF}	Fore wing pitching moment coefficient
$\bar{c}$	Global mean aerodynamic chord
C_1	Wing mass estimation coefficient
dia	Diameter
DOC	Direct operating cost
Eqn	Equation
FEA	Finite element analysis
FEM	Finite element method
FR	Fuel ratio
$FWTM$	Fore wing torsion box mass
FW	Fore wing
ft	Feet
h_n	Control fixed longitudinal location of neutral point
h_0	Aerodynamic center position on reference chord
kg	Kilogramme
kts	Knots

L_A	Aft wing lift
L_F	Fore wing lift
LFL	Landing field length
$\ln$	Natural logarithm
M_A	Aft wing pitching moment
MAC	Mean aerodynamic chord
max	Maximum
MDO	Multidisciplinary design optimization
M_F	Fore wing pitching moment
M_o	Aircraft pitching moment
m	meter
M_{TOM}	Maximum take-off mass
M_W	Wing mass
M	Aircraft design mass
MLG	Main Landing Gear
NLG	Nose Landing Gear
m	Meter
N	Normal acceleration factor
nm	Nautical mile
NP	Neutral Point
n_{ult}	Ultimate load factor
OEM	Operating empty mass
$pass$	passenger
q	Dynamic pressure
S	Gross wing area
S_A	Aft wing area
S_F	Fore wing area
sfc	Specific Fuel Consumption
S_{csw}	Control surface area
SM	Static margin
t/c	Thickness to chord ratio
TAS	True Air Speed
TFL	Takeoff field length
TO_{Dist}	Takeoff distance
USD	United States Dollar
V_D	Design diving speed
VBA	Visual Basic for Applications
$\bar{V}_T$	Tailplane volume ratio
W_{des}	Design all-up weight of aircraft
W_W	Wing group weight

Greek Letters

$\Lambda_{1/4}$	Quarter chord sweep angle
$\Lambda_{1/2}$	Mid chord sweep angle
λ	Taper ratio
τ	Average thickness to chord ratio
Λ	Wing sweep angle
α	Angle of attack
α_A	Aft wing local incidence
η	Elevator angle
β_η	Elevator trim tab angle
ε	Downwash angle
η_A	Aft wing setting angle

Subscript

A	Aft wing
B	Box
C	Cantilever
F	Fore wing
r	Wing root

Introduction

1.1 Background

Aviation has witnessed remarkable advances in the last 40 to 50 years in terms of aerospace materials, computational aerodynamics, avionics and jet engine efficiency. However, there is an on-going search for the next future airliner configuration that would bring about even more significant improvements in fuel efficiency and reductions in noise and noxious emissions⁵. This is because current civil transport aircraft configurations, exemplified by their forerunner, the Boeing 707 (Fig 1.1), seem to have reached their limit of optimization thereby renewing interests in unconventional aircraft designs.



Figure 1.1: Boeing 707 Aircraft¹

Several efforts, by governments, industry and the academia, at seeking the next future aircraft configuration are on-going. An example is the silent aircraft initiative undertaken by Cambridge University and MIT which examined the blended wing body configuration² shown in Fig 1.2. The Intergovernmental Panel on Climate Change is an example of governmental organizations which support proposals for new airframe designs that would have greater fuel efficiency⁶. It is along this approach of new airframe designs with promise of improved fuel efficiency that this work considered the box/joined wing aircraft configuration.



Figure 1.2: Blended Wing Body Configuration²

The box/joined wing aircraft configuration (Fig 1.3) along with other non-planar wing concepts, were evaluated by Munk⁷ and Prandtl⁸ in the early twentieth century. In more recent times, researchers like Wolkovitch⁹, Kroo¹⁰, Henderson¹¹ and Frediani¹² have also investigated non-planar lifting surfaces and box/joined wing aircraft configurations. The box/joined wing aircraft's claimed potentials of reduced structural weight and direct operating costs, have made it a candidate for consideration as a configuration for the future. Schneider¹³ states that a 1% reduction of drag for an airliner saves 400,000 litres of fuel and thus 5000 kg of noxious emissions per year. Therefore, the possibility of lower induced drag and improved fuel efficiency with the box/joined wing aircraft have been further reasons to investigate the configuration.

1.2 Research Aim and Objectives

The aim of this research is to produce a conceptual design and optimization methodology for box wing transport aircraft. To do this will require:

- a. Examining structural, aerodynamic, and stability and control the issues associated with box wing designs.
- b. Evaluating economic potential of box wing aircraft designs.

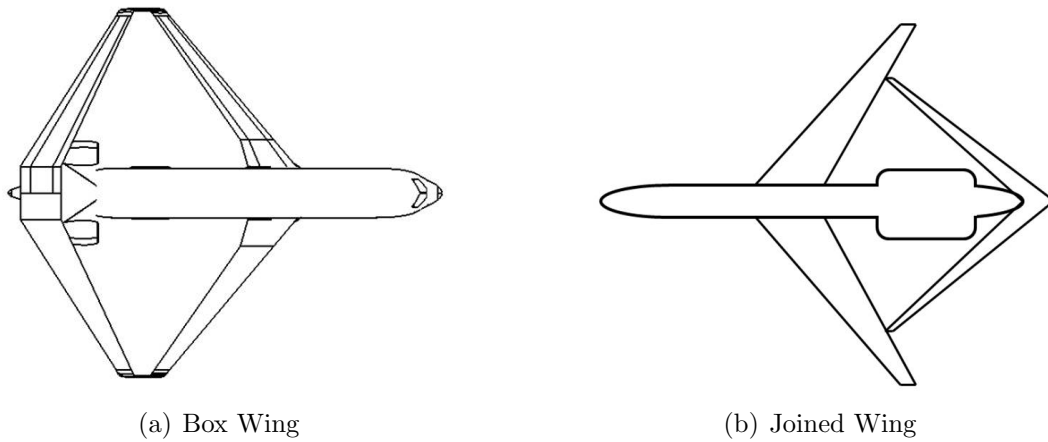


Figure 1.3: Box and Joined Wing Aircraft Configurations

To achieve this aim the following objectives were set out:

1. To develop a wing mass estimation correction coefficient for box wing aircraft.
2. To investigate box wing aircraft wing/tip fin joint fixity.
3. To investigate box wing aircraft tip fin inclination.
4. To develop a novel MDO methodology for box wing aircraft.
5. To investigate stability and control aspects of box wing aircraft.
6. To investigate economic aspects of box wing aircraft.

1.3 Thesis Structure

This thesis begins with a review of literature on joined/box-wing aircraft before presenting the wing mass estimation correction coefficient developed. The investigations on wing/tip fin joint fixity precedes the analysis of the structural effects of box wing tip fin inclination. Thereafter, case studies of MDO performed using the design/optimization method developed are presented. Stability and control investigations of the box wing are presented next before CAD implementation and the discussion chapter. The thesis ends with the conclusions and recommendations for future work.

Literature Review

2.1 Joined/Box Wing Background

Most of the literature on joined/box wing aircraft is credited to Wolkovitch⁹ but prior to this time, work was carried out on similar concepts by Munk⁷ and Prandtl⁸. However, it was Wolkovitch⁹ who published the initial extensive work on the concept and he is quoted by most researchers of the joined/box wing aircraft configuration. Wolkovitch⁹ views the joined/box wing aircraft configuration as a highly integrated concept that connects structural and aerodynamic properties in novel ways. This view is substantiated by Bernardini et al¹⁴ who also lists the advantages of joined/box wing aircraft as inclusive of light weight, high stiffness, reduced wetted area, good transonic drag, high trimmed maximum lift coefficient, low induced drag and reduced parasite drag. Furthermore, direct lift control, side-force control capability as well as good stability and control are enumerated as added attributes of the concept. As a consequence of the potentials of this configuration, a number of studies have been performed on the concept as a whole and on aspects of the design.

2.1.1 Weight Aspects

Wolkovitch⁹ states that joined wing aircraft typically weigh 65 to 78% of the weight of aerodynamically equivalent conventional airplanes. Bell¹⁵ agrees by stating that a joined wing's smaller span reduces the overall weight of the aircraft and reduces the bending moment on wings, as the length of the moment-arm is reduced. However, Wolkovitch⁹ cautions that weight is saved if the geometric parameters of the joined wing such as sweep, dihedral, taper ratio and joint location (as a fraction of span) are properly chosen. He adds that the relative advantage of the joined wing improves as wing sweep is reduced. Hajela's¹⁶ study of the sensitivity of joined wing aircraft structural weight to the wing sweep and dihedral of tip-joined wings indicates that a wing sweep of about 17 degrees produces the lightest structure.

Wolkovitch⁹ also states that weight is saved if the internal wing structure is optimized with the wing box occupying the section of the airfoil between 5% and 75% chord as sketched in Fig 2.1. He recommends the consideration that a joined/box wing's mean geometric chord

be about 65% of the conventional equivalent. He points out that the inclined plane of the joined wing will cause a forward bending moment about the vertical axis. To counter this bending moment, Wolkovitch⁹ states that the structural material distribution should be as far away from the inclined bending plane as possible. This requires that the upper leading edge and lower trailing edge of a joined wing contain the most structural material possible as also highlighted in Fig 2.1. Blair and Canfield's¹⁷ integrated design process for generating high fidelity analytical weight estimations for joined wing concepts also showed that there are weight savings for this configuration compared to equivalent conventional airplanes.

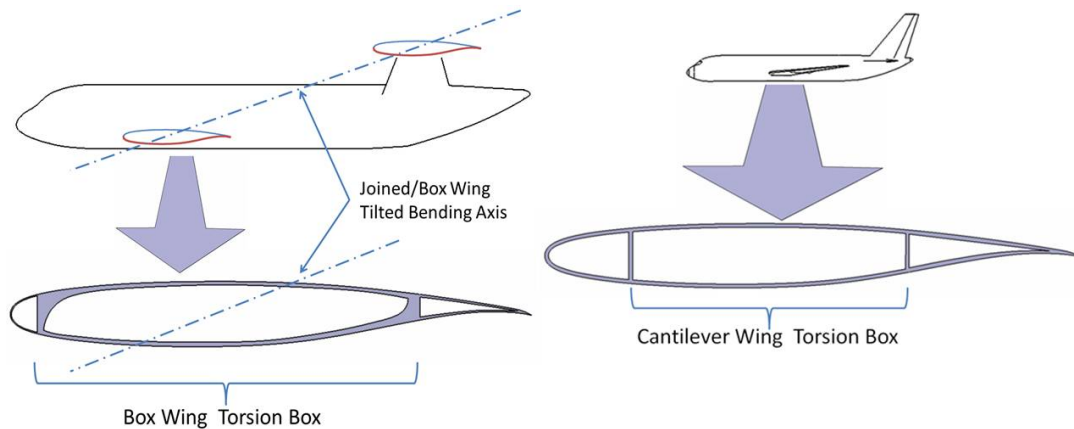


Figure 2.1: Wing Internal Structure

2.1.2 Airfoil Issues

According to Wolkovitch⁹ joined/box wings airfoils must consider the induced flow curvature and design methods similar to those used for multi-element airfoils should be employed particularly for airfoils in the vicinity of the inter-wing joint. He also recommends the use of natural laminar flow airfoils. This view is substantiated by Addoms¹⁸. Addoms¹⁸ posits that biplane configurations must employ airfoils having a substantially different camber than those of a monoplane, as the use of monoplane airfoils on biplanes causes premature separation and hence low maximum lift. Wolkovitch⁹ sums up airfoil issues by stating that the use of off-the-shelf monoplane airfoils for such configurations is disadvantageous and is no longer necessary in view of the current state of airfoil design technology. As far as the issue of stalling is concerned, Bell¹⁵ highlights that for joined/box wings the rear wing induces an upwash on the forward wing, which in turn induces a downwash on the rear wing. He states that the higher angle of attack on the front wing ensures that it always stalls first causing the nose of the aircraft to drop. By this behaviour, the joined/box wing configuration could be safer in stall than conventional aircraft.

Wolkovitch⁹ also posits that because the effective beam depth, d , of a joined/box wing is primarily determined by the chord of its airfoils as sketched in Fig 2.2, their thickness is secondary making joined/box wings suitable for thin airfoils. This means lower weight penalties than for a cantilever wing. He suggests the use of twin fins of approximately

60 degrees dihedral to reduce the unsupported column length of the aft wing. The use of twin fins for joined/box wing aircraft is subscribed to by Frediani¹². Apart from structural reasons, Frediani¹² highlights the enhanced aerodynamic efficiency given to the configuration by the aerodynamic channel defined by the top of the rear fuselage, aft wing under surface and the twin tail; although special design of this portion is required for the claimed efficiency. Bernardini et al¹⁴ developed an aerodynamic methodology for the joined/box wing configuration and concurs with Frediani¹² on the aerodynamic benefits of the aft wing/twin fin design.

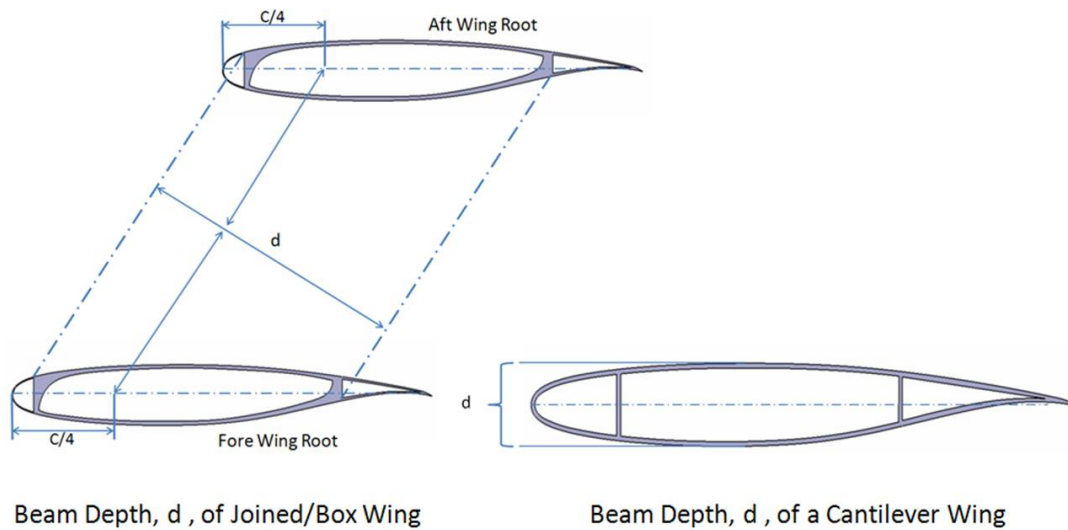


Figure 2.2: Effective Wing Depth, d

2.1.3 Structural Aspects

Wolkovitch⁹ claims that joined/box wing torsional stiffness is high because the torsion of one wing is resisted by the flexure of the other which also means higher aileron effectiveness than is obtainable with conventional wings of similar weight. This condition also means higher flutter speeds. Bagwill and Selberg¹⁹ concur by stressing the importance of having wing tip gap to wing span ratios not greater than unity in order to keep the geometry as close to a reasonable truss like structure as possible. Bagwill and Selberg²⁰ also investigated twist and cant angles of the tip fins of box wing aircraft. Their results are in accordance with Wolkovitch's⁹ results. They indicate that with certain twist and cant angles, lift to drag ratios above that of a cantilever wing configuration with higher aspect ratio than the box wing are achievable. Nangia et al's²¹ study of unconventional high aspect ratio joined wings highlight the lower induced drag of joined/box wing aircraft as well as its higher wing stiffness compared to conventional cantilever aircraft. Patil²² concurs on joined wing stiffness but he found the non-planar joined wing to be stiffer than the planar joined wing. Marisarla et al²³ undertook an investigation of the structural behaviour of the wings of a joined wing high altitude long endurance aircraft. By using loads obtained from CFD, non-linear and linear finite element analysis was performed and the difference between the values of maximum deflection obtained from linear and nonlinear analysis was only about 2%. Jemitola et al²⁴ investigated tip fin inclination effect on

structural design of a box-wing aircraft. Considering 5 tip fin inclinations the authors established that tip fin inclination significantly affects the torsional force, dragwise shear force, and dragwise bending moment distributions in the wings of a box-wing aircraft. However, for out-of-plane bending moment and shear force distributions, there were only minor variations as a function of tip fin inclination. Thus, no significant variations in wing structural design drivers as a function of tip fin inclination were observed. However, the authors recommend that flutter and divergence analysis of the box-wing configuration be performed for a more complete aeroelastic investigation into the effects of tip fin inclination on box wing aircraft.

By way of a general overview, Wolkovitch⁹ suggests that the fore and aft wing tips do not overlap in plan view as this reduces aerodynamic efficiency. This view is supported by Bagwill and Selberg¹⁹ and Frediani¹². Wolkovitch also proposes that the fore wing be located forward on the fuselage and filleted appropriately. Filleting is also recommended for the aft wing under-surface and vertical tail to minimize separation. As for wing twist Wolkovitch proposes wash-out for the fore wing and wash-in for the aft wing. He also states that it would be beneficial if the aft wing incorporates less camber than the fore wing. Bagwill and Selberg¹⁹ concur with Wolkovitch⁹ on wash-out and wash-in for the wings. Hajela and Chen²⁵ developed an approach for optimum sizing of cantilever and joined wing structures based on representing the built up finite element model of the structure by an equivalent beam model. Their method enables the rapid estimation of the optimum structural weight of wing structures for a given geometry and a qualitative description of the material distribution. Sotoudeh and Hodges²⁶ introduced a new way of analyzing statically indeterminate structures as used on joined wing aircraft. Their formulation leads to the solution of a linear system of equations at each incremental loading step, thus avoiding the numerical difficulties associated with solving nonlinear systems of equations. This includes finding suitable initial guesses needed for a NewtonRaphson solution of both statically determinate and indeterminate structures.

Cesnik and Brown²⁷ assessed the use of existing piezoelectric material technology for induced strain and producing wing-warping control on joined-wing aircraft configuration. This study was conducted based on a proposed framework which captures the nonlinear deflection behavior of the wings, the effects of anisotropic piezoelectric composites embedded in the skin, and the unsteady subsonic aerodynamic forces acting on the wing. Their study show the criticality of the sudden rear wing loss of stiffness (or buckling) that compromises vehicle integrity for joined wing configuration.

2.1.4 Interwing Joints

One of the outcomes of Wolkovitch's⁹ research is that the inter-wing joint location of 70% gives the lightest wing structure while the tip-jointed configuration is heavier than the conventional configuration of the same span. He adds that by reducing the span, the tip-jointed configuration is 20% lighter than an equivalent conventional configuration. He suggests weight minimization by decreasing the effective span/depth ratio which can be achieved by large dihedral (positive and negative), low sweep angles (positive and

negative) and high taper ratios (fore and aft). As for wing joint fixities (the attachment that connects the rear wing tips to the forward wing tips), Lin et al²⁸ examined 8 different joint types employing linear finite element method analysis as well as experimental analysis on a wind tunnel model. They conclude that the rigid wing joint has the best structural characteristics and hence is the most practical for classical joined wings shown in Fig 2.3. However, they recommend caution in attempting to extrapolate these results to other joined/box wing configurations and loading conditions. Kimler and Canfield²⁹ studied the structural design of wing twist for pitch control of a joined wing sensorcraft. The investigation involved adding a span-wise sliding joint into the wing structure of the vehicles aft wing. The joint section where the forward and aft wings connect and form the outboard wing was also redesigned and analyzed to improve the load transmission between the wing spars. Using finite element methods, their results showed that the design of the interwing joint influenced the buckling resistance of the fore wing.

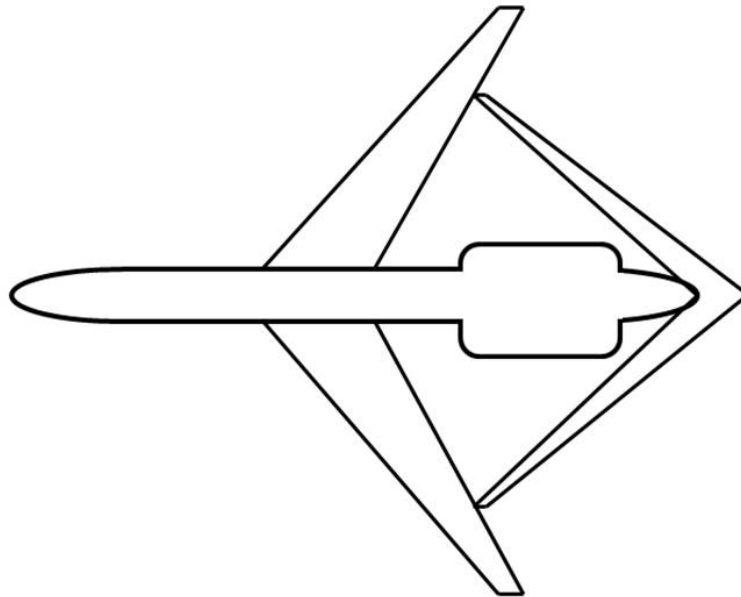


Figure 2.3: Joined Wing Configuration

2.1.5 Aerodynamic Issues

Bagwill and Selberg¹⁹ posit that positively staggered joined wings are more aerodynamically efficient than negatively staggered joined wings. By positive stagger they refer to where the higher wing is in front of a lower aft wing and by negatively staggered they refer to the reverse configuration. Mamla and Galinski³⁰ agree with Bagwill and Selberg¹⁹ on the superior aerodynamic efficiency of positive stagger joined wings over negative stagger. Smith and Stonum³¹ performed aerodynamic investigations of different joined wing aircraft configurations in a wind tunnel. Their results showed that joined wings have very good aerodynamic performance and acceptable stability and control throughout their flight envelope. Zhang et al³² developed a code to automatically generate input files for the application of CFD to joined wing aircraft design. Their study showed that a considerable amount of time required for aerodynamic analysis can be saved by their method.

Schiktanz and Scholz³³ examined the conflict of aerodynamic efficiency and static longitudinal stability of box wing aircraft. To achieve intrinsic stability in their model, the fore and aft wing lift coefficient ratio was increased in favour of the fore wing and individual centers of gravity of the airframe, engines, fuel and payload all located approximately at the same position. Demasi³⁴ investigated the conditions for minimum induced drag of closed wing systems and c-wings using lifting line theory and small perturbation acceleration potential. He solved the problems numerically and analytically and his results indicate that closed wing systems (like biplanes) have practically the same induced drag as c-wings. These results are similar to what Kroo³⁵ obtained in his investigation of non-planar wing concepts. Burkhalter et al³⁶ performed an investigation of the downwash effects for joined wing aircraft using experimental and theoretical approaches. Their results indicate differences between the experimental and the semi-empirical method of less than 12%. Corneille's³⁷ wind tunnel investigation of joined wing configurations also indicates that certain joined wing configurations outperform their conventional cantilever wing counterpart. Corneille's³⁷ findings is similar to that of Jansen et al³⁸ who performed single-discipline aerodynamic optimization and multidisciplinary aerostructural optimization investigations on nonplanar lifting surfaces. When only aerodynamics was considered, the box wing and joined wing were found to be optimal. When aerostructural optimization was performed, a winglet configuration was found to be optimal.

2.2 Configuration for this Study

Joined/box wing aircraft are called different names like box wing, biplane and diamond wing. The essential difference is in the wing configuration and the principle of operation. Suffice to state that the major difference between the two aircraft configurations is that whereas for the box wing both wings produce equal amounts of lift⁸, for the classical joined wing the fore wing produces about 80% of the total lift. Frediani's¹² research on the joined/box wing aircraft configuration draws from Prandtl's⁸ *'best wing system'* in which he showed that a closed rectangular lifting system would produce *'the smallest possible induced resistance for a given span and height'*. Prandtl⁸ established that all biplanes have less induced drag than the equivalent monoplane when the spans are equal and that biplane drag decreases as wing gap increases. Frediani¹² posits that Prandtl's⁸ *'best wing system'* if applied to current aircraft could offer induced drag reductions of up to 20-30% based on a wing gap/span (h/b)ratio of 10-15%. Frediani's¹² states that for a box wing or PrandtlPlane as he calls it, the aerodynamic efficiency is strongly linked to stability of flight and the challenge is to obtain a stable aircraft with equal lift on both wings which is the condition for Prandtl's *'best wing system'*. Frediani states that induced drag accounts for approximately 43% of the total aircraft drag during cruise in still air. Thus, if it is reduced significantly it would translate into benefits like reduced aircraft weight and reduced thrust requirements, which could mean reduced negative impact on the environment. Already, 1:5 scaled model of Frediani's PrandtlPlane has been flight tested with results that are in accordance with the mathematical model of the aircraft³⁹.

The joined/box wing configuration studied in this research is based on Prandtl theory for drag reduction of multi-plane aircraft and is therefore more inclined to the PrandtlPlane¹² than the classical joined wing emphasized by Wolkovitch⁹.

2.2.1 Background Theory

Prandtl's theory shows that the total vortex induced drag, D_i , of an unstaggered biplane's wings (Fig 2.4) can be written as

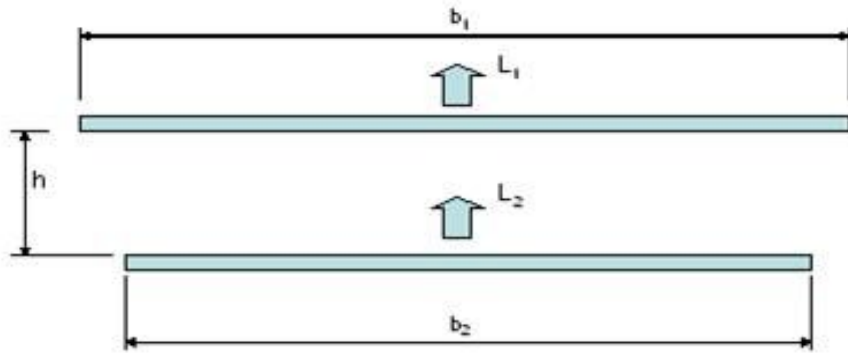


Figure 2.4: Biplane Wing Schematic

$$D_i = \frac{2L_1^2}{b_1^2 V^2 \rho \pi} + \frac{2L_2^2}{b_2^2 V^2 \rho \pi} + \frac{4L_1 L_2 \sigma}{b_1 b_2 V^2 \rho \pi} \quad (2.2.1)$$

The third term represents the mutual interference between the trailing vortices of each wing. The factor, sigma (σ), is a function of the vertical displacement, h , between the wings while b_1 and b_2 are the spans (Fig 2.4).

For a given total lift

$$L = L_1 + L_2 \quad (2.2.2)$$

the induced drag is a minimum when

$$\frac{L_1}{L_2} = \frac{b_1(b_1 - b_2 \sigma)}{b_2(b_2 - b_1 \sigma)} \quad (2.2.3)$$

For circumstances when $b_1 = b_2$ and consequently $L_1 = L_2$, the minimum induced drag, D_{imin} , can be expressed as

$$D_{imin} = \frac{2L^2}{b^2 V^2 \rho \pi} \times \frac{(1 + \sigma)}{2} \quad (2.2.4)$$

In coefficient form it becomes

$$C_{D_{imin}} = \frac{C_L^2}{A\pi} \times \frac{1 + \sigma}{2} \quad (2.2.5)$$

where A is the total aspect ratio of the lifting system.

It is evident from Equation 2.2.5 that if a biplane has the same aspect ratio and lift coefficient as a monoplane, the induced drag is reduced by the factor

$$K = (1 + \sigma)/2 \quad (2.2.6)$$

According to Prandtl, the ideal arrangement for minimum induced drag is a closed biplane with same lift distribution and same total lift on each of the wings. In this closed biplane the upper portion of the endplates is subject to outward pressure and the lower portion to inward pressure. Fig 2.5 shows a front view schematic of 2 equal span lifting surfaces joined at the tips with the ideal pressure distribution on the end plates. As the gap between the wings increases the load distributions vary and become more uniform and the system becomes more efficient. This is because the forces on the end plates are effective in reducing the trailing vortices, which would appear at the wing tips if they were not joined.

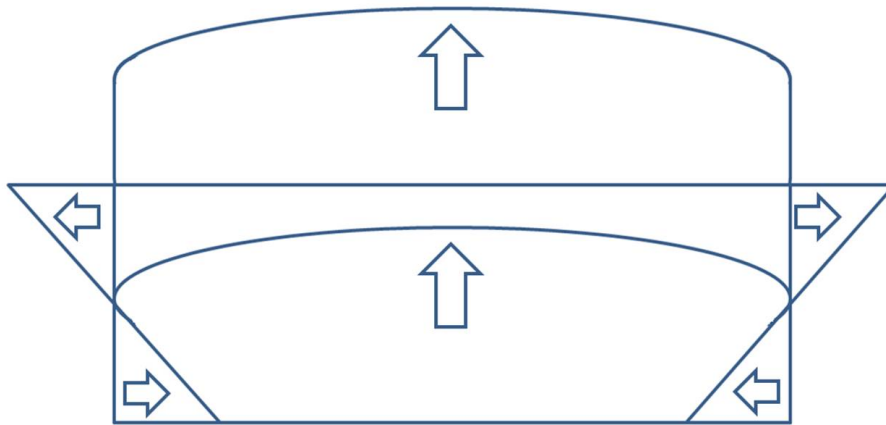


Figure 2.5: Biplane Lift Distribution

A plot of Prandtl's induced drag reduction theory (Fig 2.6) shows the decrease in induced drag that can be obtained with the closed biplane arrangement. For example, for a h/b

of 0.25, the induced drag is about 71 percent of an equivalent monoplane with the same aspect ratio. However, the upper limit of the benefits of this trend would be determined by wing mass increase and practicability of the design

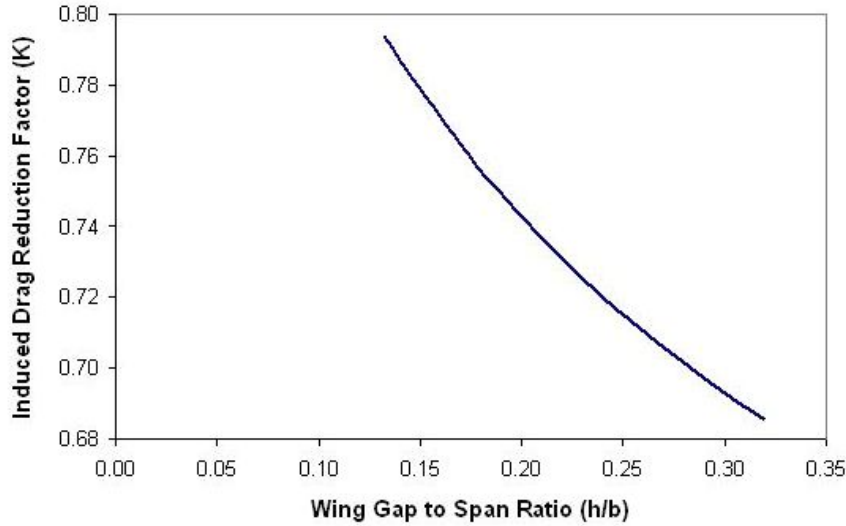


Figure 2.6: Effect of Wing Gap on Induced Drag Reduction

Using Munk's equivalence theorem, Prandtl's theory, which is for an unstaggered biplane can be extended to staggered wing arrangement. The theorem states that the total induced drag of any multiplane system remains unaltered if any of the lifting elements are moved in the direction of motion provided the lift distribution remains constant. However, by staggering the wings, the induced flow between the wings automatically changes. The forward wing experiences an up wash while the aft wing is subject to a downwash field due to the fore plane. This means that the lift-curve slope of the aft wing will be less than the fore wing for identical airfoil sections and angles of attack (assuming no fuselage is present) as already explained by Addoms¹⁸. One of the challenges of the box wing is therefore to optimize the design such that the Prandtl condition i.e. equal lift on both wings is achieved.

2.3 Joined/Box Wing Aircraft Optimization

The potentials of joined/box wing above conventional aircraft have been well highlighted and as mentioned Section 2.1, these potentials are only if configuration parameters are carefully selected. Hence, a number of optimizations have been performed on the concept; Livne⁴⁰ advocates the use of a multidisciplinary design approach to simultaneously design for aerodynamics and structures due to the complex interactions that joined wing configurations create. This study integrates structural and aerodynamic design issues of box wing aircraft into a single design optimization process.

Gallman et al⁴¹ examined a joined wing configuration using a numerical optimization method. A vortex-lattice model of the complete aircraft was used to estimate aerodynamic performance and a beam model of lifting surface structure was used to estimate wing and tail weight. The intentions of the study were to show the application of numerical optimization to aircraft design and to present a quantitative comparison of joined wing and conventional aircraft designed for the same medium-range transport mission. They also developed a computer program capable of modelling joined wing aircraft and estimating their overall performance in terms of DOC. Aerodynamic forces were calculated using a LinAir program in which a vortex lattice model of the complete aircraft was built. The structural design algorithm considered one manoeuvre load case, several gust load cases and the nonlinear effects of secondary bending moments. A numerical optimizer (NPSOL) was used to design the joined wing and a conventional transport with minimum DOC.

Gallman et al⁴¹ note that buckling is a design issue for some joined wing structures, especially '*Wolkovitch Joined Wings*'. They posit that joined wing aircraft have structural efficiency but poor high-lift capability and that it suffers a substantial penalty in maximum lift because of a short tail moment arm and the corresponding large tail download required to trim in takeoff configuration. They propose moving the mass centre of the empty aircraft forward, thereby reducing the operational range of centre of mass and designing to a lower level of static stability as ways to reducing the takeoff field length. They also state that redistributing the fuel and locating the engines on the wings could have a significant impact on takeoff field length and the resulting effect on overall performance could make the joined wing DOC less than that of the conventional configuration. They state that an extra fuel tank placed in the tail and used to trim at the aft-most center of gravity, has the most significant impact on joined wing performance.

Gallman et al⁴¹ also opine that an in-depth study of wing sweep, flap span and elevator span might show further improvements in joined wing performance and that any design changes that reduces tail sweep is likely to improve joined wing performance. They posit that takeoff field length and horizontal-tail buckling represent the most critical design constraints for joined wings. They added that a significant increase in *DOC* is caused by the joined wing's inability to generate maximum lift in takeoff configuration. They conclude that joined wing that carry payloads that allow for a reduction in tail sweep and that reduce the influence of tail download on maximum lift may perform better than conventional configuration aircraft. The joined wing's considered by Gallman et al was a '*Wolkovitch Joined Wing*' different from that of this study which is based on Prandtl's '*best wing system*' and is a box wing.

Rasmussen et al⁴² considered the optimization process for a flexible joined wing. The intention was to locate optimal '*regions and design trends of the joined wing aircraft considering six configuration variables*'. The research investigated parametric configuration changes that were optimized for flexible static air loads. Structural optimization was carried out using MSC.NASTRAN and aerodynamic loads that were generated using

MSC.FlightLoads. Weight optimization was achieved by varying spar, rib and skin thicknesses of the wing structure. A response surface was created from 78 samples acquired from the optimization process. Classical minimization techniques were applied to the response surface and only three unique optimal points were found all of which were reanalyzed to establish their weight values. Rasmussen et al conclude that the response surface created was valid for some general trends and finding optimal regions and that a higher number of samples would be required to refine the response surface appropriately. Again the joined wing considered by Rasmussen et al was a '*Wolkovitch Joined Wing*' different from that of this study which is based on Prandtl '*best wing system*' and is a box wing.

Gallman et al⁴³ performed a synthesis and optimization for a medium range joined wing transport aircraft. They developed a program to model joined wing transports and estimate their overall performance in terms of DOC. The program predicted the aerodynamic interaction between the '*lifting surfaces and the stresses in the statically-indeterminate structure*'. Aerodynamic forces were predicted using a vortex lattice model of the complete aircraft in a LinAir program. Viscosity and compressibility were added to compute compressibility drag and inextensible theory was used to design fully stressed lifting surface structures. Manoeuvre load case and several gust cases, as well as the effects of secondary bending moments were considered in the design algorithm. Tail buckling was investigated using secondary bending moments. The weight computations were combined with a statistically based method from Douglas Aircraft Company to obtain an estimate of lifting surface total weight. A numerical optimiser, NPSOL, was used to design the joined wing for minimum DOC. The optimization problem consisted of 11 design variables and 9 constraints. The outcome was that the joined wing is cheaper to operate than an equivalent conventional transport especially if fuel prices were high and that the aircraft is deficient in the field performance owing to a low maximum lift capability. According to the researchers, this drawback is caused by a relatively short tail moment arm and could be ameliorated by either a larger wing area or engine or even a combination of both.

Muira et al⁴⁴ conducted a parametric weight evaluation of joined wings by structural optimization. The study used a structural weight minimization capability for arbitrary wing configurations based on a Programming Structural Synthesis System developed by the National Aeronautics and Space Administration. The purpose was to present trends of the structural weight of joined wings for conceptual/preliminary design of medium sized aircraft instead of the absolute weight of the wings. Over 50 cases of structural weight minimizations were performed. This showed that the characteristics depended strongly on wing geometry and structural arrangement. Their results show that tip joined wings were approximately 36% lighter than an equivalent cantilever wing.

Yechout et al⁴⁵ embarked on an aerodynamic evaluation and optimization of a Houck joined wing concept model aircraft. They used general engineering rules of thumb and a University of Missouri study on biplane design to optimize the joined wings performance. The authors considered changing the negative decalage angle, the a taper ratio to less than one, increased gap, decreased wing sweep and/or decreased stagger. They establish that

a gap of 4.75in with -1.5 decalage angle was the optimum design configuration producing higher lift coefficients and a more shallow drag polar. However, they state that none of the Houck joined wing configurations evaluated show significant potential for performance improvement over a monoplane.

Kapania et al⁴⁶ conducted a multidisciplinary design optimization of a strut-braced wing transonic transport. The aircraft's specification was to transport 324 passenger 7500 nautical miles at Mach 0.85. The intention was to examine the interdependency of structures, aerodynamics and propulsion vis-a-vis wing bending load alleviation due to the strut and what benefits increased aspect ratio and reduced wing thickness would accrue. They used a Virginia Tech MDO code to model aerodynamics, structures/weights, performance and stability and control. The code's primary analysis module included aerodynamics, wing bending material weight, total aircraft weight, stability and control, propulsion, flight performance and field performance. The outcome of their investigation was that the strut-braced wing has significant weight reductions with improved fuel consumption even with a smaller engine compared to a traditional cantilever configuration.

Canfield et al⁴⁷ proposed an integrated design method for joined wing configurations by using a geometric model and a user interface all using an adaptive modeling language. The model could be analyzed for structural or aerodynamic characteristics through external software. They conclude that nonlinear structural analysis is important to accurately capture the large deformations that occur on a joined wing configuration. Nangia and Palmer⁴⁸ analyze the effects of forward swept outboard wings on a joined wing aircraft. They found that a forward swept outboard wing produces favorable lift distribution on the forward and aft wing through a forward placement of the center of pressure for the overall aircraft vehicle.

Weisshaar and Lee⁴⁹ explored the configuration changes of a joined wing aircraft with respect to flutter speed using Rayleigh-Ritz modeling, composite tailoring, and optimization for a linear model. The most noteworthy results are related to joint location and sweep angle. They examined sweep angles from 30 to 45 using parametric methods. They found that as the sweep angle rose for a fixed span size the flutter dynamic pressure increased and as the joint location moved closer to the tip of the wing the flutter dynamic pressure decreased slightly.

Kroo et al¹⁰ undertook a research program aimed at improving techniques for multidisciplinary design and optimization of large-scale conventional aeronautical systems. The research involved the simplification and decomposition of analysis using compatibility constraints. The new structure, that involved coupling optimization and analysis, was intended to improve efficiency while simplifying the composition of multidisciplinary, computation-intensive design problems involving many analysis disciplines and design variables. Work was in two areas: system decomposition using compatibility constraints to simplify the analysis structure and take advantage of coarse-grained parallelism and

collaborative optimization, which is a decomposition of the optimization process to permit parallel design and to simplify interdisciplinary communication requirements. When system decomposition using compatibility constraints was applied to an aircraft design problem using PASS (an aircraft synthesis code) combined with NPSOL (a numerical optimizer) significant reductions in computational time was observed. A similar result was observed when collaborative optimization was applied to an aircraft design problem. Kroo et al's joined wing was the '*Wolkovitch Joined Wing*'.

In view of the foregoing, several optimization methods have been applied to joined wing aircraft concepts and even fewer on box wing configurations. However, none of these researchers considered the box wing aircraft concept based on Prandtl's '*best wing system*' except Frediani¹². Frediani¹² conducted a box wing aircraft design using a non-commercial MDO code called MAGIC that uses a sequential unconstrained minimization technique. The MDO code and procedure, which was conducted by 5 universities, was elaborate and computationally extensive. However, it was deficient as far as impact on the environment is concerned and did not have a structural optimization algorithm integral to it. The optimization algorithm developed for this study incorporates a structural module which is its major difference from Frediani's¹² work

2.4 Overview of Multidisciplinary Design Optimization (MDO)

According to Korte⁵⁰ '*MDO is a methodology for the design of complex engineering systems and subsystems that coherently exploits the synergism of mutually interacting phenomena . MDO permits optimization of a number of design variables affecting different disciplines, which when applied to aircraft, should eventually result in reduced acquisition and operating costs and/or better system performance*'. Typical Engineering optimization problems can be classified as constrained optimization and unconstrained optimization problems. Constrained optimization is the minimization/maximization of an objective function subject to restrictions on the responses and/or objective function. These constraints can be either equality or inequality constraints. On the other hand, unconstrained optimization problem is not subject to any restriction on the values of the responses and/or objective function. In addition, unconstrained optimization is central to the development of most optimization algorithms. This is because constrained optimization algorithms are often an extension of unconstrained optimization. Most optimization algorithms are iterative and can be represented by the equation:

$$X^q = X^{q-1} + \alpha^q S^q \text{ }^{50}$$

where q is the number of iterations, S is the vector search direction, α is the scalar direction and the initial solution X^0 is given. MDO usually begins with the selection of design variables, constraints, objectives and models of the various disciplines. This process is

called problem formulation and its major constituents are outlined below.

Design Variables

Design variables are specifications that are controlled by the designer. They are terms that represent physical features of the design that will be parametrically varied to achieve a desired effect or find an ‘ideal’ set of features. Design variables can be continuous such as wing areas, discrete such as number of engines or number of aisles and seats across or Boolean such as whether to build a monoplane or a biplane. Furthermore, it is common for the design variable to be confined within the bounds of maximum and minimum values. Depending on the solution method, these bounds can be treated as constraints. The full range of design variables is often called the ‘Design Space’⁵¹.

Constraints

Constraints are must-meet capabilities or conditions that the design is bound to satisfy in order for it to be physically realizable or acceptable and occur due to the finiteness of resources or technological limitation. They impose limits on the responses and/or objective functions. Design constraints can be performance constraints such as takeoff distance, rate of climb, or cruise speed or cutoff values for physical features such as wingspan or fuselage diameter. Design constraints can also include environmental restrictions such as noise propagation. Constraints can be expressed explicitly in the solution algorithm or incorporated into the objective function using Lagrange multipliers⁵¹.

Objective or Measure of Merit (MOM)

The objective or measure of merit (MOM) is the desired capability or characteristics that the vehicle will be optimized to attain. Many optimization techniques work only on single objective; other methods allow multiobjective optimization, such as the calculation of a Pareto front. When using these methods it is usual to weigh the various objectives and sum them to form a single objective^{50 52}.

Parameters

Parameters are quantities that affect the objective but are considered fixed in that they cannot be changed by the designer. Sometimes parameters can be turned into design variables to enlarge the design space. A parameter could also be a former design variable that was fixed at some values because they were found not to affect any objective or because their optimal level was predetermined⁵¹.

Model

A model is an object that has the ability to predict the behaviour of a real system under a set of defined operating conditions and simplifying assumptions. It is an empirical or theoretical function that relates the constraints and objectives to the design variables⁵⁰.

2.4.1 Types of MDO Algorithms

There are 2 types of optimization algorithms for solving multidisciplinary optimization problems. These are the deterministic gradient search algorithm and the stochastic non-gradient based algorithms. Within these, the choice of an optimization technique depends on factors like number and type of design variables, whether the problem is a linear or non linear problem and constraints (equality/inequality or unconstraint).

Stochastic methods are easy to program, do not require continuity in problem definition (hence ideally suited to discrete and or combinatorial type of optimization problems) and can make use of a large number of processors. Nevertheless, it requires more function evaluations to find an optimum solution as compared to the gradient based algorithm. Gradient based algorithms are known for their inability to cope with multiple minima and their inefficiency when dealing with a large number of design variables. Furthermore, the gradient based algorithms are subject to numerical noises which can cause incorrect gradients. This can delay or prevent convergence. However, gradient based approach is more advantageous because it is deterministic, fast and results in lower computational cost⁵¹.

2.4.2 Merits and Demerits of MDO

The merits of MDO include reduction in design time, a systematic logical design procedure, the fact that it handles a wide variety of design variables and constraints and it is not biased by experience. The disadvantages include the fact that computational time grows with number of design variables, numerical problems increase with number of design variables and it can be difficult in dealing with discontinues functions.

2.5 Multidisciplinary Design Optimization Setup for Study

For this study, the following are the MDO setup

Constraints - Takeoff distance, cruise speed, range and landing distance.

Parameters - Number of engines, fuselage diameter, aerofoils, wing span, wing gap.

Design Variables - Wing sweep, wing area and average thickness to chord ratio.

Objective Function (Measure of Merit) - Minimization of all up mass, fuel per passenger per nautical mile and direct operating cost per nautical mile.

2.6 Chapter Summary

Several authors have established the potentials of joined/box wing above conventional cantilever wing aircraft. These potentials have inspired various optimization studies to be performed on the configurations which have brought to light the unique challenges of the joined/box wing configuration. As for box wing aircraft there is a gap in knowledge with respect to an MDO setup that has an integral structural optimization module. Accordingly, the MDO framework of this study, outlined in the next chapter, integrates in one suite all the major issues of box wing aircraft design/optimization.

Methodology Framework

This chapter introduces the way in which the optimization methodology for the study was developed before outlining the baseline models used for the research. It also presents the major design and analysis tools used.

3.1 Design/Optimization Tool

A design/optimization tool was developed for this study. The design tool was implemented in Microsoft Excel as it is a ubiquitous tool that can be enhanced by Visual Basic for Application (VBA) algorithms. The tool was setup to solve multiobjective and multidisciplinary optimization problems using deterministic gradient search and stochastic non-gradient search algorithms. The architecture of the design tool is as shown in Fig 3.1 and it consists broadly of baseline design, geometry definition, wing structures, mass, aerodynamics, performance and cost modules. The arrows show the direction and paths of optimization routines of the tool. The tool was designed for box wing optimization but was also modified and used for optimizing the baseline conventional cantilever wing aircraft.

In the tool's MDO setup the constraints are takeoff distance, cruise speed and landing distance while the parameters are number of engines, fuselage diameter, aerofoils and span. The design variables are wing sweep, thickness to chord ratio, wing area and wing gap while the objective functions or measures of merit are minimization of all up mass, fuel per passenger per nautical mile and DOC per nautical mile.

As indicated by the arrows in Fig 3.1, initial design variables from the baseline designs were inputted to the geometry definition module. These inputs were then fed into the wing structures and aerodynamic modules. The aerodynamic module generated loads which were inputted to the wing structures module. The outputs of the wing structures module were the wing masses which were fed into the mass module. The outputs of the aerodynamics module were lift and drag forces both of which were inputted to the performance module. The performance module also got weight inputs from the mass module. Local iterations were also performed between the mass, aerodynamics and performance modules to satisfy the objective functions. Based on the computations in the performance module cost optimization was performed and stability and control investigated. The pro-

cess was iterated until minimum all up mass, fuel per passenger per nautical mile and DOC per nautical mile was achieved.

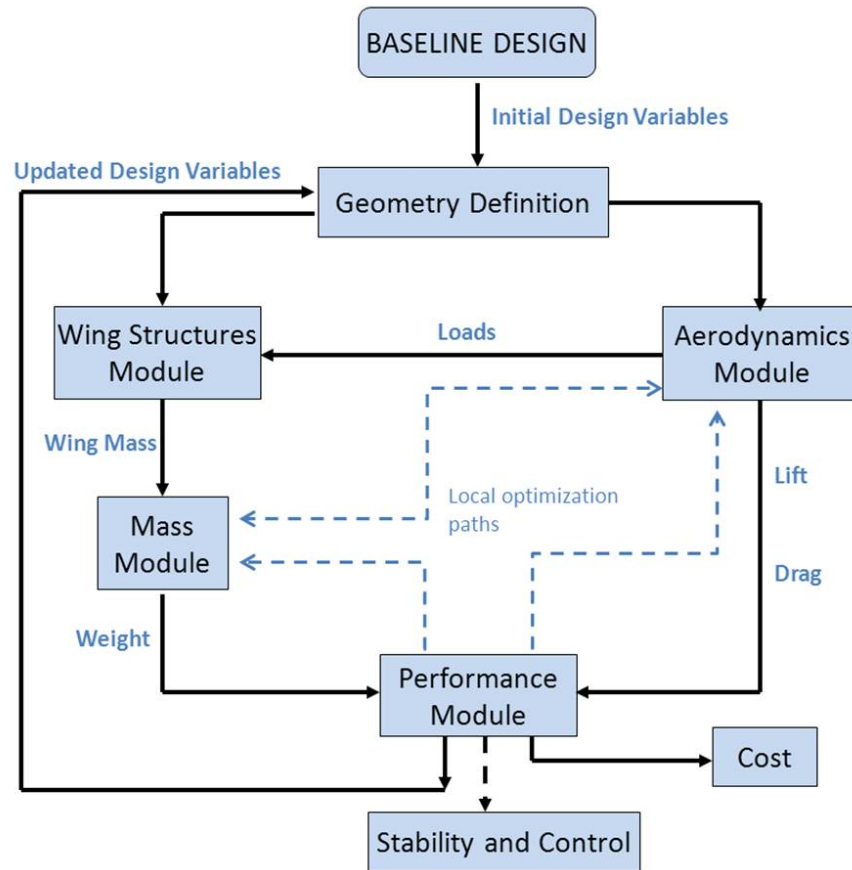


Figure 3.1: Design Tool Architecture

3.1.1 Wing Structures Module

The wing structures module used inputs from the geometry definition module and loads from the aerodynamics module to estimate the wing masses. Off line finite element analysis was used to determine the appropriate algorithms for estimating the wing mass of the box wing aircraft. This was then embedded in the wing structures module details are in Chapter 4.

3.1.2 Mass Module

The mass module is expanded in Fig 3.2 and consists of the zero fuel mass and the fuel mass sub modules. The zero fuel mass sub module consist of the operating empty mass and payload routines. The operating empty mass routine is broken down into the fuselage, nacelle and power plant, tail fin, landing gear, tip fin, fixed equipment, wing mass and

control surfaces subroutines. Note that the operating empty mass routine gets wing mass inputs from the wing structures module. The wing mass is in turn influenced by the wing sweep angle, wing gap and thickness to chord ratio subroutines. All the modules and sub modules incorporate empirical mass estimation formulae from Howe⁵³ and Jenkinson⁵⁴. These modules are iterated to arrive at the optimum design using codes written in VBA. Two codes were written for the wing structures and mass modules. Macro one called RangeGSeek performs optimizations for 741 aircraft samples which are all modifications of the baseline box wing aircraft in accordance with the MDO outlined in Chapter 2.5. Macro two called SelectAUM stochastically selects samples out of the outcome of Macro one using predetermined cut-off parameters. The parameters are fore and aft wing sweep angles. Details of the RangeGSeek and SelectAUM macros one and two are in Appendix A.

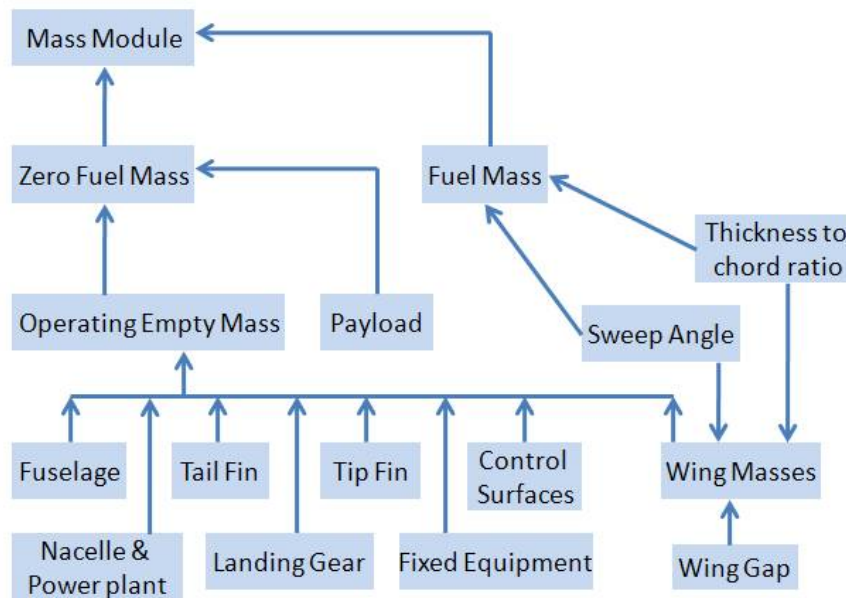


Figure 3.2: Mass Module Architecture

3.1.3 Aerodynamics Module

The Aerodynamics module shown in Fig 3.3 consists of the lift loads and drag sub modules. The lift loads module uses an off line vortex lattice tool called Athena Vortex Lattice (AVL)⁵⁵, to generate the lift loads on the wings. The drag sub module consist of the zero lift (consisting friction, form, and interference drags) and induced drag routines. The induced drag routine is a function of the wing gap subroutine. The Zero lift drag routine computes the drag of components like fuselage, nacelle, engine pylons, landing gear blisters, tail fins and tip fins using methods outlined in Hoerner⁵⁶, Roskam⁵⁷ and Jenkinson et al⁵⁴. A factor to account for wing shield, secondary items and wave drag is also included. For all components their individual Reynolds numbers, skin friction coefficient, form factor, interference factor and wetted area were computed and summed

up. These along with the drag from the wings as a function of wing sweep (wing sweep subroutine) and thickness to chord (thickness to chord subroutine) ratios are summed up to get the overall sample drag. Zero lift drag optimization as a function of sweep angle is performed using a VBA macro called SelectCDo (see Appendix A). Induced drag inputs are taken from AVL.

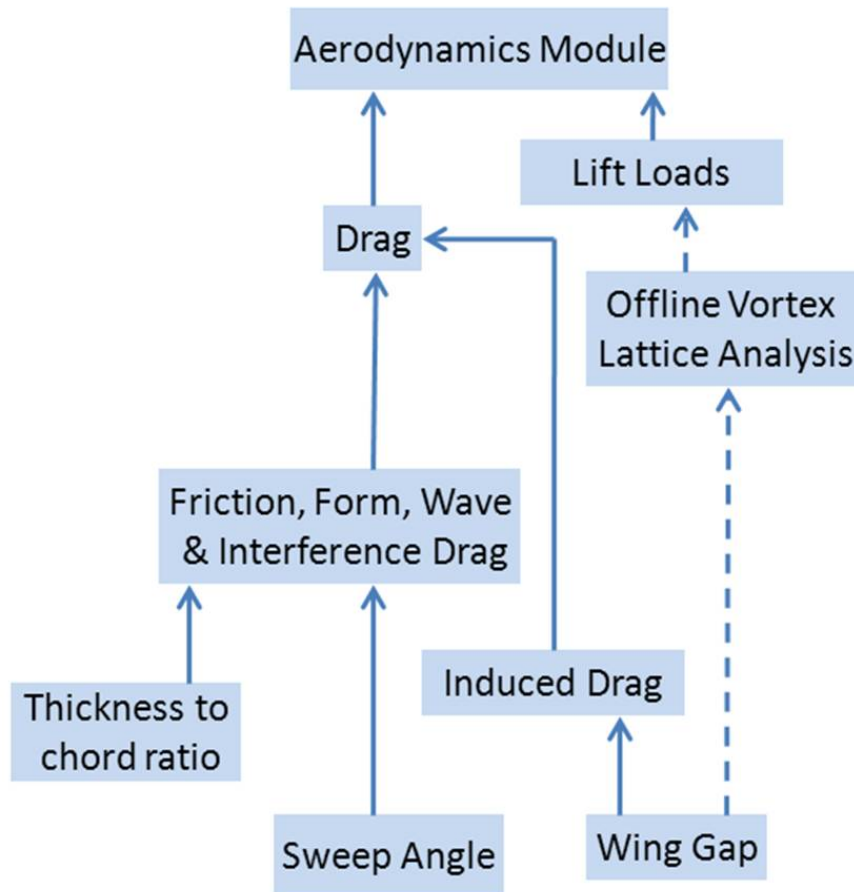


Figure 3.3: Aerodynamics Module Architecture

In Fig 3.3 the variables were thickness to chord ratio, sweep angle and wing gap. As the optimization routines were run within the module, outputs of lift and drag forces as a function of the variables were fed into the performance module. Similarly, aerodynamic loads were output to the wing structures module.

3.1.4 Performance Module

The performance module is shown in Fig 3.4 and consists of the field, flight and stability and control sub-modules. The field performance sub-module contains routines for computing the field performance of selected outcomes from the mass and aerodynamic modules using the sequence and method in Jenkinson⁵⁴. The flight performance sub-module evaluates the flight performance using the Breguet range equation and plots the

payload range graph. The stability and control sub-module uses an off-line software to perform trim, short period oscillation and phugoid analysis.

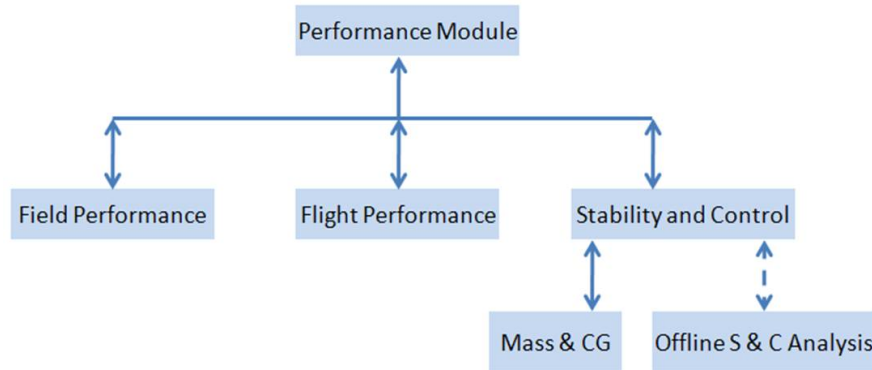


Figure 3.4: Performance Module Architecture

3.1.5 Cost Module

The cost module shown in Fig 3.5 consist of the DOC, fuel per pax per nm, DOC per nm per seat and aircraft market price sub-modules. All sub-modules use the sequence and method outlined in Roskam⁵⁸. The cost modules optimization is performed using a VBA code called AcCost (see Appendix A). The code performs an iterative sequence between the sub-modules to present the DOC, fuel per pax per nm, DOC per nm per seat and aircraft market price for all aircraft models in the envelope.

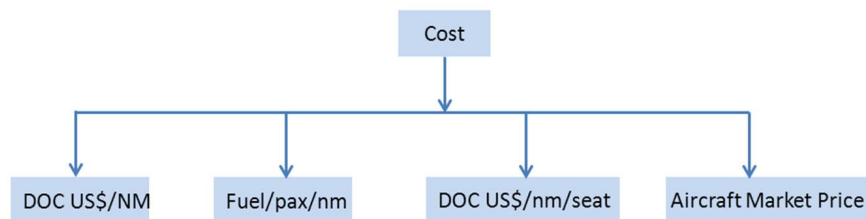


Figure 3.5: Cost Module Architecture

3.1.6 Stability and Control Module

The stability and control module shown in Fig 3.6 consist of the trim analysis, phugoid analysis and short period analysis submodules. It is off-line and gets inputs from the performance module. As the names suggest trim, phugoid and short period oscillation analysis were performed in each module. The phugoid and short period analysis were performed only on models that could be trimmed. Further information on the stability and control tool used is in Section 3.7.

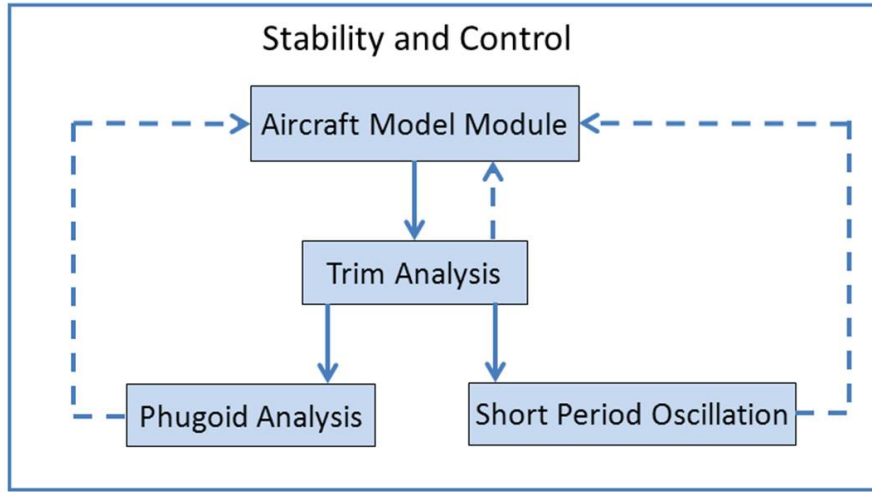


Figure 3.6: Stability and Control Module Architecture

In Fig 3.6 mass, inertia and aerodynamic characteristics of each aircraft model was inputted from the performance module. These inputs were used to build models in the aircraft model module. Trim analyses were then performed on each model and models that were not trimable were discarded. Phugoid and short period oscillation analyses were then performed on the models that were trimable. If the models satisfied the requirements given in Moorhouse and Woodcock⁵⁹ it was then inputted in the pilot opinion chart⁴. If the model did not satisfy the requirements the model was modified and the process repeated again.

3.2 Baseline Medium Range Box Wing Aircraft

The baseline aircraft of study is taken from Smith and Jemitola⁶⁰ which is derived from Jemitola⁶¹ and shown in Fig 3.7. It is a 4000 nautical mile range box wing airliner with a maximum takeoff mass of 127760kg and wing span of 37.6m. The fore and aft wing gross areas are $118.32m^2$ each, while the sweep angles are 40 and -25 degrees respectively. Overall fuselage length is 46 meters and maximum diameter 5.6m. A background to the baseline aircraft and other aircraft parameters are in Appendix B.

3.3 Baseline Conventional Cantilever Aircraft

For comparison purposes a baseline conventional cantilever aircraft, Fig 3.8, was also obtained from Jemitola⁶¹. It is also a 4000 nautical mile range airliner but with a maximum takeoff mass of 136000kg and wing span of 47.0m. The wing gross area is the same as the sum of the fore and aft wings of the baseline box wing aircraft at $236.64m^2$, while the wing sweep angle is 30 degrees. Overall fuselage length is 46 meters and maximum diameter 5.6m. Other aircraft parameters are in Appendix B.

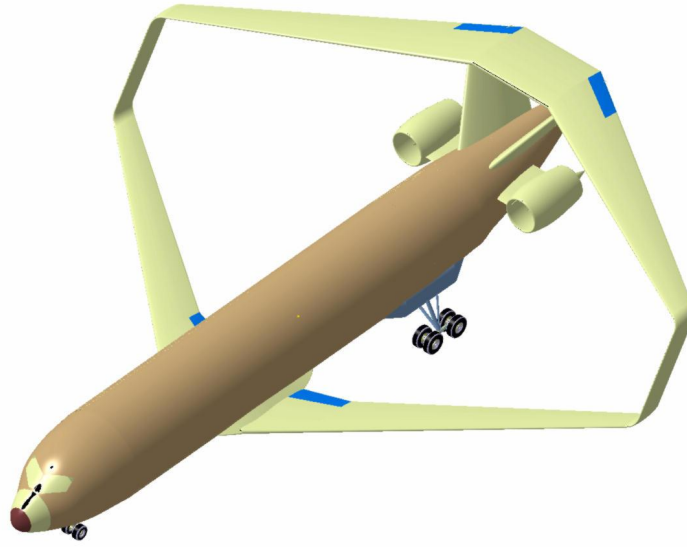


Figure 3.7: Baseline Medium Range Box Wing Aircraft

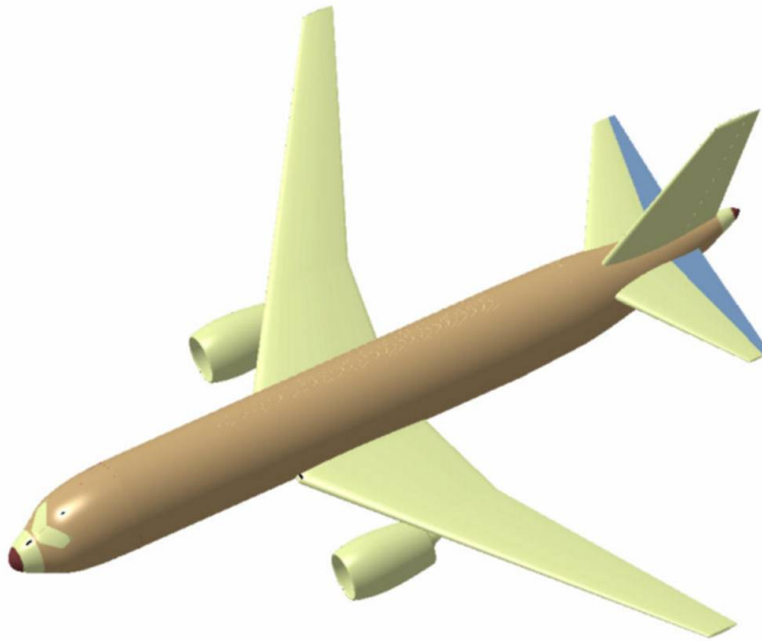


Figure 3.8: Baseline Medium Range Box Wing Aircraft

3.4 Vortex Lattice Tool

Aerodynamic loads for all models in this study were generated using a vortex lattice software called Athena Vortex Lattice (AVL)⁵⁵. Vortex Lattice Methods have been used for box wing configurations and it showed good accuracy in analyzing the configuration⁶². AVL is a program that utilizes vortex-lattice theory for aerodynamic and dynamic analysis

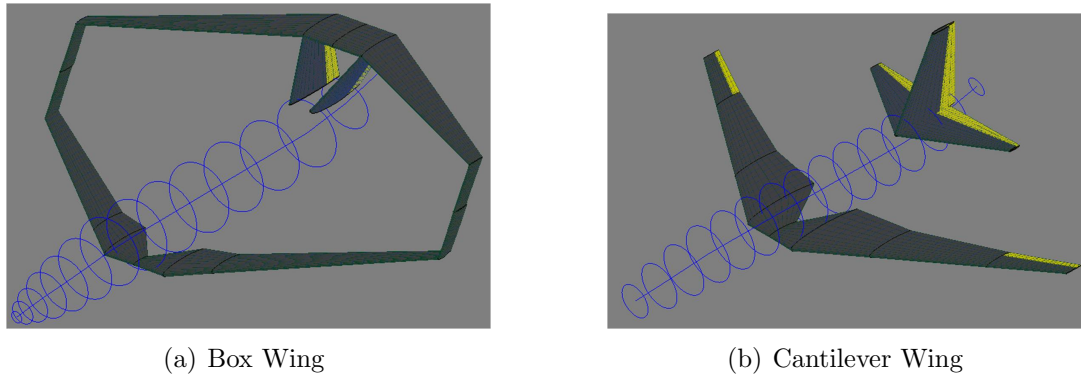


Figure 3.9: AVL Models of Baseline Aircraft

of a given aircraft geometry. It uses a vortex lattice model for the lifting surfaces, together with a slender-body model for fuselages and by simulating the flow field, it predicts the pressure distribution around the simulated body. To create a model, AVL requires a text file that defines aircraft geometry. The aircraft is defined as a series of sections running parallel to the chord line. The text file also contains wing and tail surfaces defined in Cartesian coordinates. Sections are needed at root, tip and each end of the wing section or control surface. AVL connects the leading and trailing edges of these sections to create a complete model. In Fig 3.9 are AVL models for both baseline aircraft. The fuselage's effect on the lift distribution was accounted for by implementing 40% of the wing root chord at the center line of the model as given in Howe⁶³ and shown in Fig 3.9. The AVL model text files for the box wing and conventional cantilever wing aircraft are in Appendix C

3.5 Aerodynamic Loads

Figure 3.10 shows the load distributions for the box and cantilever wings obtained from AVL. The load distributions included the contributions to torsion coming from the airfoil section properties and the moment arm between the lift application point and the elastic axis of the torsion box. The assumed flight condition was a 2.5g manoeuvre case limit load at sea level at design diving speed, V_D , at maximum take-off weight⁶⁴. In a final design analysis, other loading conditions should also be examined, which include various possible combinations of control deflections and gust loadings. However, the conditions that were assumed in this preliminary study are considered adequate.

3.6 Finite Element Analysis Tool

Strand7⁶⁵ was used for the finite element analysis of this study. Several different Strand7 elements could be used to model an aircraft wing. Only beam, plate and brick elements were considered for the wing finite element models. The beam element was chosen because

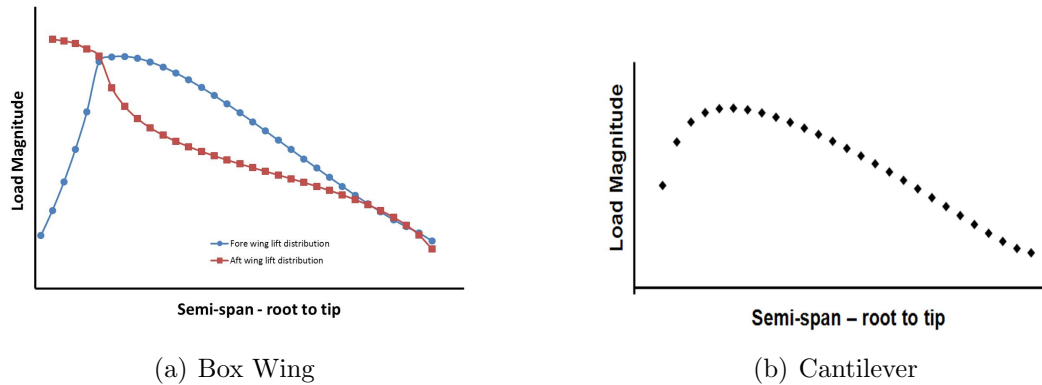


Figure 3.10: Wing Load Distributions

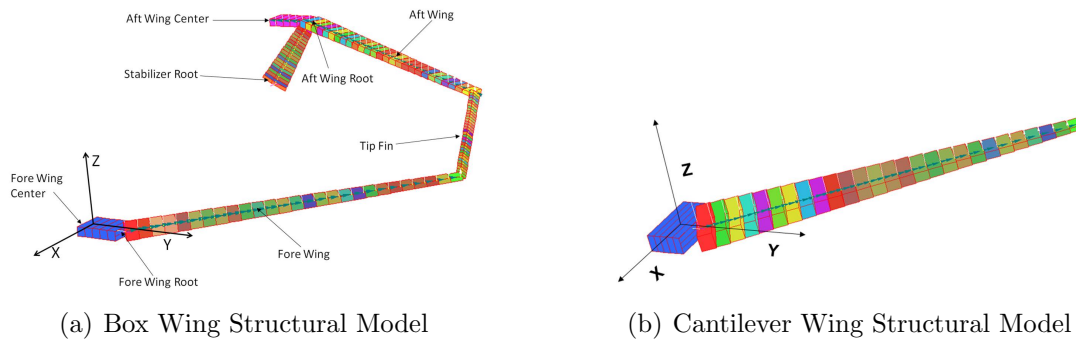


Figure 3.11: Strand7 Wing Structural Models

of the enormous gain in simplicity over a plate or brick model. Beam is a generic name for a group of one-dimensional or line elements. These elements are all connected between two nodes at their ends and the single dimension is length. Other dimensions can be specified from a range of beam sections available thereby giving 3-dimensional properties. In its most general form the beam element can carry axial force, shear force, bending moment and torque. The beam element can also be used to simulate a bar with a non-uniform cross section by interpolating sections. Strand7 has the capability of applying distributed or concentrated loads to a beam element. It also can apply distributed or point torque or moment on beam elements. Fig 3.11 presents a beam version of the box and conventional cantilever wing models. The global coordinate system used for the static analysis is also defined in Fig 3.11. The beam version of the box wing configuration contains 119 elements while the cantilever wing has 32 elements.

3.7 Wing Cross Section Properties

Studies by Wolkovitch⁹ indicate that the optimum wing torsion box cross-sectional profile of the box wing configuration is one which accounts for the tilted bending axis of the wings by having the '*bending-resistant material*' concentrated near the upper leading edges and lower trailing edges. However, for simplicity, an idealized wing torsion box cross-sectional

geometry, illustrated in Fig 3.12, was used. Although this is not the optimum torsion box cross section for a box wing, it was assumed to be adequate for the study. Furthermore, since the wing cross sections are perpendicular to the wing's elastic axis, the root and tip of the swept wings create modeling problems. However, for slender swept wings and applied loads through the wing elastic axis, an effective wing root and tip may be assumed, simplifying the wing finite element model.

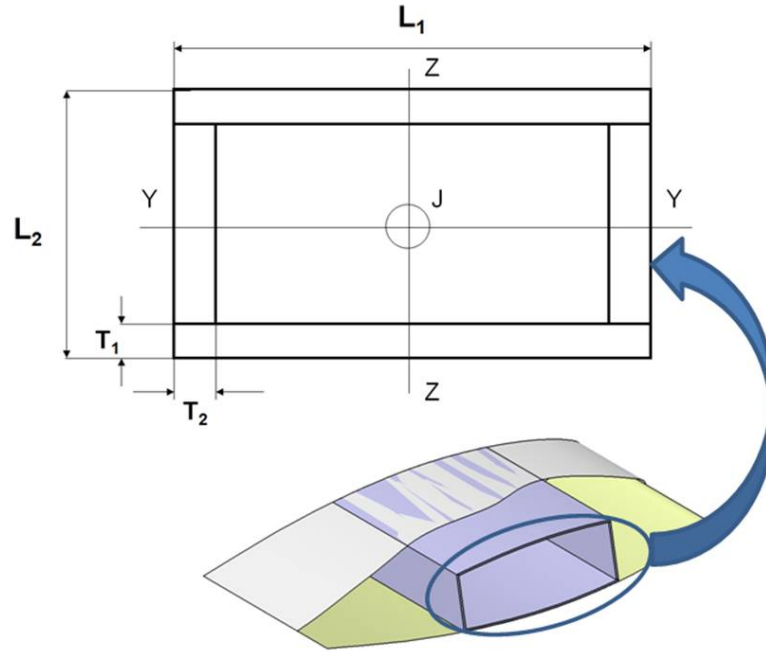


Figure 3.12: Idealized Torsion Box Cross-Section Geometry

The specific airfoil sections used in this study were not required due to the conceptual nature of the analysis although Wolkovitch⁹ has highlighted the suitability of box wings for thin airfoils. In addition, the torsion boxes for all finite element models were assumed to occupy 50% of the chord. The torsion box properties implemented on the model were that of the baseline model obtained from Smith and Jemitola⁶⁰ and are detailed in Table 3.1.

Table 3.1: Torsion Box Properties

Item	$L_1(\text{m})$	$L_2(\text{m})$
Aft Wing Center	1.84	0.44
Aft Wing Root	1.84	0.44
Tip Fin	0.93	0.22
Fore Wing Root	2.00	0.48
Fore Wing Center	2.00	0.48
Stabilizer Root	2.48	0.6

Strand7⁶⁵ beam elements require cross sectional properties as program inputs and automatically compute properties such as wing cross sectional area A , the second moments of

area I_{yy} and I_{zz} , and the polar second moment of area, J . Note that for a symmetrical cross section, $I_{yz} = 0$. The material properties used for all the finite elements are those of Aluminium alloy 2024 - T42 selected from the Strand7 material database; modulus of elasticity $E = 73.0\text{GPa}$, modulus of rigidity $G = 27.6\text{GPa}$, density $\rho = 2770\text{kg}/\text{m}^3$.

3.8 Structural Sizing and FEA Procedure

Structural sizing was done by implementing the algorithms outlined in Howe⁶³. Due to the simplicity of the torque box and elastic axis beam type models used for the stress/strain analysis, only general stress trends were analyzed. These trends were determined as a function of the beam span as opposed to a plane stress or shear flow analysis which would also estimate the stresses around the torque box. Stiffness criteria that ensure adequate wing aero-elastic performance was accomplished following Howe's⁶³ approach. The idealized wing structural box model used for the finite element analysis makes the distributed flange construction method ideal for sizing the torsion box skin and web, T_1 and T_2 in Fig 3.12.

As the analysis in this study is conceptual the fuselage is omitted from the idealization but accounted for by the boundary conditions imposed on the model. The boundary conditions used are shown in Table 3.2. Wing inertia relief was ignored because its presence or absence was not going to affect the trends observed. In addition, the analysis was conducted with cases in iso-inertia conditions (equal to zero). Due to the symmetric nature of the torsion box model, the elastic axis and geometric centres of the wing models coincide and it was along this axis that the loads were applied. For this reason forces and moments are transmitted through the wing tip joints for the box wing models (a single grid point) by equating the respective translations and rotations of the joint receivers at the joint.

Table 3.2: Wing Mass Estimation FEM Model Boundary Conditions

Axis	Fore Wing Center	Wing Root	Aft Wing Center	Stabilizer Root
Translation				
x	Free	Fixed	Free	Fixed
y	Fixed	Fixed	Fixed	Fixed
z	Free	Fixed	Free	Fixed
Rotation				
x	Fixed	Free	Fixed	Fixed
y	Free	Fixed	Free	Fixed
z	Fixed	Free	Fixed	Fixed

The torsional loads from the airfoil section properties and from the moment arm between the lift application point and the elastic axis of the torsion box were combined to generate an equivalent torque applied along the elastic (and geometrical) axis of the torsion box. By superposition principle, said equivalent torque and the lift load (applied along the

elastic axis as well) were considered together to achieve a representative FEA model of the physical system.

The structural sizing and FEA procedure for each model followed the method outlined in Fig 3.13. Initial pooling values for T_1 and T_2 (see Fig 3.12) were implemented in Strand7⁶⁵ elements database. Thereafter a 3-dimensional model of the port wing torsion box assembly was built in Strand7⁶⁵. The aerodynamic loads obtained from AVL⁵⁵ were then applied in a distributed manner along the elastic axis of the wing model and a linear analysis performed. The results of this linear analysis were then used to compute new values of T_1 and T_2 . If there was no change between the old and new T_1 and T_2 values, the model had converged. If not, the new T_1 and T_2 values were implemented in the Strand7⁶⁵ elements database and a new FEM model of the same wing was built. Aerodynamic loads were then applied on this new model and a linear analysis performed. The results were subsequently used to compute another set of T_1 and T_2 values. If the newly computed values were the same as the previous T_1 and T_2 values, that model represented the converged model of the wing. If not, the process was repeated until there was convergence. This procedure was performed for all wing models used in this study. The shear, torque and bending moments outputs for all the converged solutions were obtained and compared.

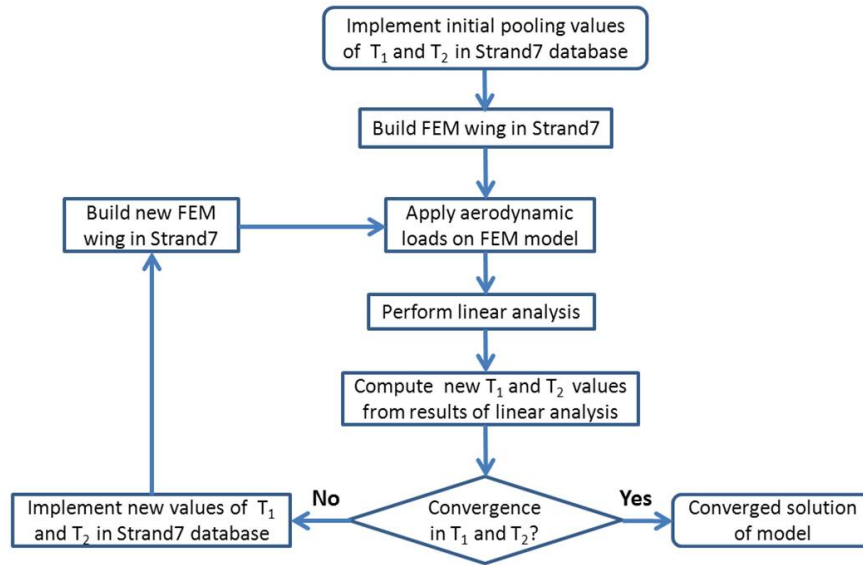


Figure 3.13: Structural Sizing Procedure Schematic

3.9 Stability and Control Tool

Trim, short period oscillation and phugoid stability analysis in this study was performed using J2Universal software suite⁶⁶. The software is a tool kit that can amongst others be used to investigate an aircraft's stability and control characteristics. The package consists

of 5 sub-elements:

J2 Builder which is a graphical interface used to develop aircraft models; aircraft component positions, mass inertia are entered as a starting point. Aerodynamic forces and moments are generated off line and also entered into the software. J2 Builder then constructs the aircraft model which can be used for assessment.

J2 Freedom provides flight dynamics simulation of aircraft models, allowing evaluation of the flight envelope and enabling trim and response scenarios to be created.

J2 Visualize also provides understanding and evaluation of aircraft behaviour through data visualization and graphic displays. J2 Visualize enables the creation of graphs and traces with which to view the data either as an analysis is underway or as a post processing tool. The system enables the establishment of optimum configurations by plotting graphs from the evaluated data and by viewing and monitoring individual analysis as they happen.

J2 Virtual allows the viewing of the aircraft in a virtual 3 dimensional environment to understand what happens during manoeuvres and gives the user a clear understanding of how the aircraft will respond to commands.

J2 Elements utilizes integrated strip theory to calculate total aerodynamic coefficients and derivatives automatically.

Although J2 is a versatile tool for investigating stability and control is it possible to get erroneous results. The source of errors has to do mostly with the inputs into the tool; wrong aerodynamic and engine data inputs would produce faulty results. Thus, it is important to ensure that input data is as accurate as possible. Another limitation of J2 is that the aerodynamic computations do not consider span-wise aerodynamic interactions which means that induced drag computations may not be very accurate. Furthermore, J2 only computes downwash at the aft lifting surface upwash is not computed hence for box wing configuration. This has to be accounted for by the aerodynamic inputs for the fore wing. Lastly, J2's algorithms are almost entirely based on work by Roskam⁵⁸ and so may not be able to accurately model box wing configurations. However, in the absence of more suitable and capable software J2 suffices and the outputs for the box wing can be cross-checked with predictions and results of other researchers for validity.

3.10 Chapter Summary

The baseline box wing and conventional cantilever wing aircraft for this study have been introduced. The architecture for design/optimization procedures, including the sub-

modules, used to refine the designs have been presented. Furthermore, the off-line vortex lattice method, finite element method and aircraft dynamic analysis tools used have been elucidated. These procedures and tools were subsequently used to perform structural and aerodynamic investigations of the baseline aircraft beginning with wing mass estimation algorithm for box wing aircraft.

Wing Mass Estimation Algorithm

4.1 Introduction

One of the challenges in the conceptual design of a box wing airliner is estimating the wing mass. Several empirical formulae exist for estimating the mass of conventional cantilever wings but these would be misleading if applied directly to an unconventional configuration like box wing aircraft. Anderson and Udin⁶⁷ present a theoretical wing mass derivation formula for subsonic aircraft. The formula breaks the wing into an inboard, midboard and outboard wings and by an analysis of the wings loads the relative masses of the wing components are computed as a function of the bending moment and shear forces. The formula also requires the geometric characteristic of the wing as well as '*manufacturing coefficients*' as inputs. The results show accuracy within 10 percent and root mean square of 6 percent. However, the details required in using the equation precludes its use at the conceptual stage of any design. Furthermore, it cannot be applied on box wing aircraft as it is based on conventional cantilever wings.

The study by Miura et al⁴⁴ on parametric weight evaluation of joined wings by structural optimization was aimed at presenting trends and not development of a wing mass estimation algorithm. Furthermore, the trends presented cannot be applied to box wing configurations as their study relates to classical joined wing aircraft only. Similarly, Blair and Canfield's¹⁷ development of an integrated design process for generating high fidelity analytical weight estimations is for classical joined wings and so do not apply to box wings.

Howe⁶⁸ states the importance of accurate wing mass prediction in determining an optimum aircraft design but admits that lack of and inconsistency of data, definition of what constitutes the wing mass and material properties are the main difficulty in mass prediction methods. He presents an empirical wing mass prediction method called the C_1 method where mass is calculated as a function of the main geometric and operational parameters but states that some discretion is needed to account for any special features of the design. C_1 is a coefficient dependent on the type of aircraft. Howe⁶⁹ also presents a theoretically based approach called the F method where the wing mass is expressed as a sum of the mass of the span wise covers/booms and shear web of the structure, the mass

of the ribs, the mass of high lift devices and secondary fairings and the mass of miscellaneous items such as attachments for stores, power plants and landing gear. Both methods show good results but while the empirical method is straightforward in application, it is slightly less accurate than the theoretical method. The F method is inappropriate for use in early conceptual design stages as the details required are usually unavailable. Again these methods are for conventional cantilever wing aircraft and not box wing aircraft.

Wing mass estimation algorithms are especially useful when quick approximate answers are needed during preliminary investigations; such answers indicate trends without the necessity of detailed knowledge of the aircraft. For the box wing aircraft there are no empirical formulae for estimating wing mass. An aircraft's mass needs to be reasonably estimated at the conceptual design phase for several reasons including the direct relationship between an aircraft's operating empty mass, the aircraft's price and direct operation cost. This relationship is even more paramount for novel designs where conventional mass estimation methods may not apply. A practical approach for developing a formula for use at the early conceptual design stage would be to modify an existing algorithm by determining the appropriate coefficients for the fore and aft wings. To be of use such an algorithm must include all the variable parameters which influence the answer and must not be too complex or require detailed knowledge of the design. Torenbeek's⁷⁰ method, Eqn 4.1.1, requires many factors not available at the early conceptual design stage

$$W_{W_{basic}} = Const \times k_{no} k_{\lambda} k_e k_{uc} k_{st} \times \left[k_b n_{ult} (W_{des} - 0.8 W_w) \right]^{0.55} \times b^{1.675} (t/c)_r^{-0.45} (Cos \Lambda_{1/2})^{-1.325} \quad (4.1.1)$$

where:

k_{no} represents the factor of weight penalties due to skin joints, non-tapered skin, minimum gauge, etc.

k_{λ} represents the factor of wing taper ratio

k_e represents the factor of bending moment relief due to engine and nacelle installation

k_{uc} represents the correction factor for undercarriage installation

k_{st} represents the factor required to provide stiffness against flutter

k_b represents the factor for strut location on braced wings

Jenkinson et al's⁵⁴ method shown in Eqn 4.1.2 also demands factors not available at the early conceptual design stage as well.

$$M_W = 0.021265 \left(M_{TOM} \times n_{ult} \right)^{0.4843} \times S^{0.7819} \times AR^{0.993} \times (1 + \lambda)^{0.4} \times \frac{(1 - R/M_{TOM})^{0.4}}{\Lambda_{1/4} \times (t/c)^{0.4}} \quad (4.1.2)$$

R is the effect of inertia relief on the wing root bending moment given by

$$R = \{M_W + M_F + [(2 \times M_{eng} \times B_{IE}/0.4b)] + [(2 \times M_{eng} \times B_{OE}/0.4b)]\} \quad (4.1.3)$$

where:

M_F is mission fuel

M_{eng} is individual engine and nacelle mass

B_{IE} is distance between inboard engines

B_{OE} is distance between outboard engines

The main drawback with Raymer's⁷¹ method at the early design stage is that it requires knowledge of the wing control surface area; an unlikely information at the conceptual design stage.

$$W_W = 0.0051 \left(W_{des} n_{ult} \right)^{0.557} S^{0.649} AR^{0.5} (t/c)_{root}^{-0.4} (1 + \lambda)^{-0.1} \times (Cos \wedge_{1/4})^{-1.0} S_{csw}^{0.1} \quad (4.1.4)$$

In contrast, Eqn 4.1.5⁷² requires parameters usually available at the early design stage. Eqn 4.1.5 is based on Howe's⁶⁸ equation and he states its accuracy as 86%; a value which can be considered acceptable for the conceptual design stage.

$$M_W = C_1 \left[\frac{bS}{Cos \wedge_{1/4}} \left(\frac{1 + 2\lambda}{3 + 3\lambda} \right) \left(\frac{M_{TOM} N}{S} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \quad (4.1.5)$$

Eqn 4.1.5 is a consequence of the analysis of one hundred aircraft of all types. Its derivation involved computing the optimum minimum mass of the wings primary structure using equations based on the theoretically required bending and torsional strength and modifying this mass by making allowances for departure of the structural concept from the optimum. The component of the wing mass which ensures adequate torsional stiffness and prevents flutter is substantiated by the design diving speed in the equation.

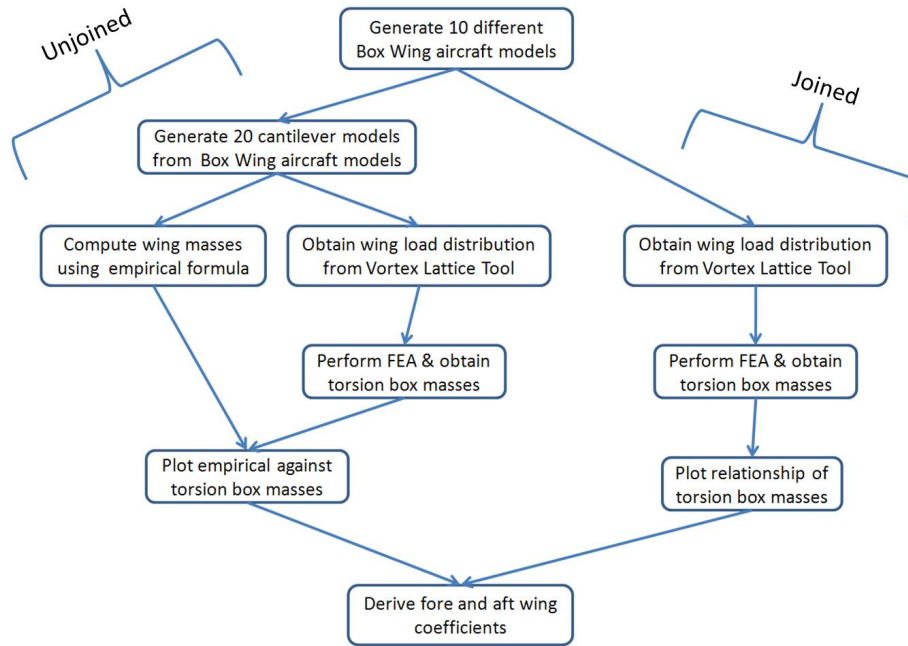
Furthermore, the mass of secondary items such as high lift devices and controls were estimated using statistical data. The algorithm is intrinsically a compromise between wing stiffness and lightness and is based on the use of aluminium alloy construction. There isn't an inertia relief factor in the equation but it is implicitly accounted for in the values of the coefficient C_1 . The algorithm also uses C_1 to account for different types of aircraft and layout details therefore suiting the purpose for a box wing estimation algorithm. Some values of C_1 given in the Aircraft Mass Prediction⁷² manual are shown in Table 4.1. What has to be determined is the value of C_1 that would apply to the fore wing and the C_1 value for the aft wing of a medium range box wing aircraft.

Table 4.1: Aircraft Type Mass Coefficients

Aircraft Type	C_1
Long Range	0.028
Short Range	0.034
Braced Wing	0.021
Light aircraft	0.028 - 0.034

4.2 Procedure

The procedure is outlined in Fig 4.1. Ten different box wing aircraft models with appropriate medium range wing parameters were generated. The wing parameters of the ten box wing models were subsequently used to generate twenty cantilever winged (ten forward swept and ten aft swept) aircraft models. The masses of these cantilever wings were then estimated using Eqn 4.1.5. Next, each of the twenty cantilever winged aircraft was modelled in a vortex lattice tool called AVL⁵⁵ (described in Chapter 3 paragraph 4) to obtain the wing load distributions for an assumed flight condition.

**Figure 4.1:** Procedure Schematic

The wing loads were then used to perform FEA on the torsion box models of the entire cantilever winged aircraft from which the torsion box masses were obtained. A relationship was subsequently established between the empirical and torsion box masses of the twenty cantilever wing aircraft. In a similar manner the ten box wing models were modelled in a vortex lattice tool to obtain the wing loads and distribution for an assumed flight condition. The wing loads were then used to perform FEA on the torsion box models of

the box wing aircraft from which their masses were obtained. A relationship was thereafter plotted with the equivalent cantilever torsion box model mass and that of the box wing. From these plots the coefficients for the fore and aft wings were derived.

4.3 Aircraft Model Parameters

For the box wing aircraft models $M_{TOM} = 127,760$ kg and for the cantilever aircraft $M_{TOM} = 63,880$ kg. All models were designed for a 3500 nautical mile range. Other model parameters are shown in Figs 4.2 and 4.3.

Model	S (m^2)	λ	b (m)	τ	Fore Wing Sweep ($^\circ$)	Aft Wing Sweep ($^\circ$)
Box 1	112	0.27	32	0.09	30	-19
Box 2	117	0.26	33	0.09	31	-20
Box 3	110	0.25	33	0.09	32	-22
Box 4	115	0.28	33	0.10	33	-24
Box 5	120	0.27	34	0.11	30	-26
Box 6	122	0.26	34	0.11	33	-28
Box 7	118	0.25	35	0.12	35	-18
Box 8	116	0.25	34	0.12	33	-20
Box 9	124	0.25	35	0.13	32	-22
Box 10	119	0.25	34	0.13	29	-24

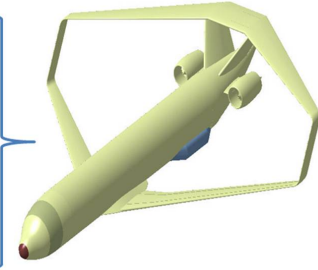


Figure 4.2: Box Wing Model Parameters

Model	S (m^2)	λ	b (m)	τ	Sweep ($^\circ$)	Model	S (m^2)	λ	b (m)	τ	Sweep ($^\circ$)
AS 1	112	0.27	32	0.09	30	FS 1	112	0.27	32	0.09	-19
AS 2	117	0.26	33	0.10	31	FS 2	117	0.26	33	0.09	-20
AS 3	110	0.25	33	0.11	32	FS 3	110	0.25	33	0.09	-22
AS 4	115	0.28	33	0.12	33	FS 4	115	0.28	33	0.10	-24
AS 5	120	0.27	34	0.10	30	FS 5	120	0.27	34	0.11	-26
AS 6	122	0.26	34	0.11	33	FS 6	122	0.26	34	0.11	-28
AS 7	118	0.25	35	0.12	35	FS 7	118	0.25	35	0.12	-18
AS 8	116	0.25	34	0.12	33	FS 8	116	0.25	34	0.12	-20
AS 9	124	0.25	35	0.13	32	FS 9	124	0.25	35	0.13	-22
AS 10	119	0.25	34	0.13	29	FS 10	119	0.25	34	0.13	-24

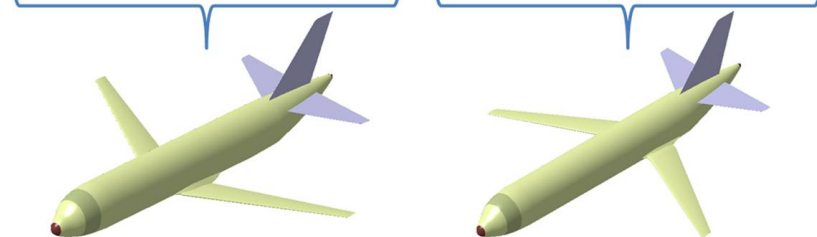


Figure 4.3: Cantilever Wing Model Parameters

4.4 Aerodynamic Loads and Finite Element Analysis

The aerodynamic loads used to derive the wing mass estimation algorithm is as explained in Chapter 3 Section 5. Similarly, the finite element analysis and structural sizing procedure performed is as outlined in Chapter 3 Sections 6, 7 and 8.

4.5 Results

The results of the outlined procedure were used to plot graphs and perform a regression analysis and their R^2 and p-values determined. R^2 is the proportion of variability in a data set that is accounted for by a statistical model⁷³. It provides a measure of how well future outcomes are likely to be predicted by a model. The P value answers the question: If there were no linear relationship between X and Y overall, what is the probability that randomly selected points would result in a regression line as far from horizontal (or further) with respect to what you observed?⁷⁴. In other words the p-value gives the probability of obtaining a test statistic as large as the one calculated from data, if in fact the true slope is zero. Thus, the smaller the p-value the more significant the regression. Customarily, if the p-value is below 0.05 the regression is significant⁷⁵. Plots of the torsion box masses of the cantilever wings and their equivalent box wing aircraft wing are shown in Figs 4.4 and 4.5. Fig 4.4, which is a plot of the cantilever wing against the equivalent fore wing of the box wing aircraft model gives a coefficient of determination R^2 of 0.9938 and a p-value of $p = 4.00 \times 10^{-10}$ showing that the regression is significant. For the aft wing, analysis gives a coefficient of determination R^2 of 0.9960 and a p-value of $p = 7.05 \times 10^{-10}$ (see Fig 4.5), again showing a high linear correlation. Thus, the relationship between the wings as cantilevers and as part of the box wing arrangement is significant.

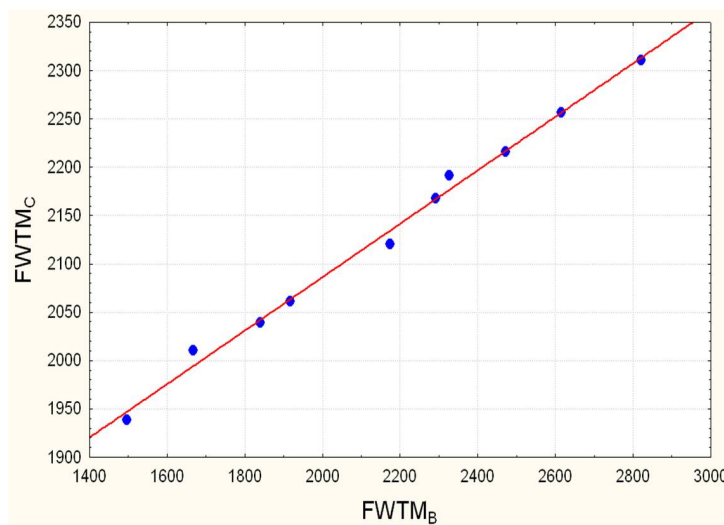


Figure 4.4: Cantilever/Box Relationship - Fore Wing

The empirical masses of the cantilever wings were computed using Eqn 4.1.5. A plot of

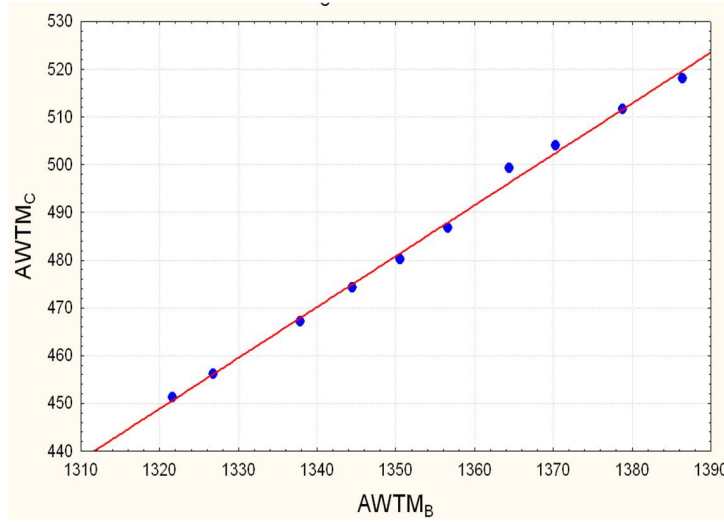


Figure 4.5: Cantilever/Box Relationship - Aft Wing

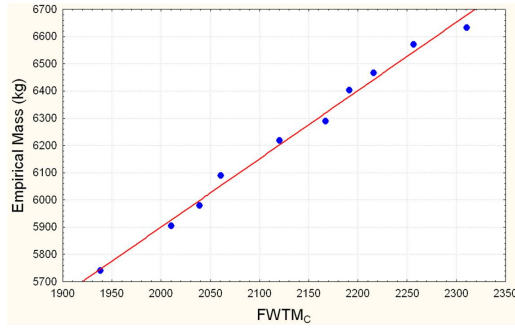


Figure 4.6: Empirical/FEA Relationship - Fore Wing

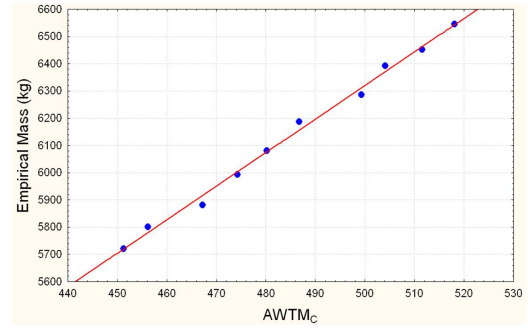


Figure 4.7: Empirical/FEA Relationship - Aft Wing

these empirical masses against their finite element model torsion box masses for the fore wing gave the plot in Fig 4.6. The analysis gives a coefficient of determination R^2 of 0.9907 and a p-value of $p = 2.09 \times 10^{-9}$ showing the high linear correlation.

The above procedure for the aft wings is shown in Fig 4.7. The regression has a coefficient of determination R^2 of 0.9946 and a p-value of $p = 2.33 \times 10^{-10}$ showing that the regression fits the points very well. It also means each individual variable has a high linear correlation with the dependent variable. The quality indicators for all analyses are shown in Table 4.2.

4.6 Coefficient Derivation

The key to developing a coefficient for estimating box wing aircraft wing mass lies in establishing a relationship between the empirical masses of the cantilever wings and the results of the FEA carried out. To this end, a relationship was established between the mass of the torsion boxes of the wings as part of a box wing arrangement ($FWTM_B$) and

Table 4.2: Regression Quality Indicators

Analysis	p value	R^2	Adjusted R^2
Fig 4.7	4.00×10^{-10}	0.9938	0.9931
Fig 4.8	7.05×10^{-10}	0.9960	0.9955
Fig 4.9	2.09×10^{-9}	0.9907	0.9895
Fig 4.10	2.33×10^{-10}	0.9946	0.9939

the mass of the wings as cantilevers ($FWTM_C$). This relationship for the fore wing is given as:

$$FWTM_C = 0.2765FWTM_B + 1533 \quad (4.6.1)$$

The relationship between the empirical mass of the fore wing as a cantilever and its torsion box mass is given as:

$$M_W = 2.5159FWTM_C + 870.96 \quad (4.6.2)$$

Thus, by substitution, the empirical mass of the wing as part of the box wing arrangement is given by:

$$M_W = 0.6956FWTM_B + 4727.83 \quad (4.6.3)$$

Relating this equation to Eqn 4.1.5 gives:

$$0.6956FWTM_B + 4727.83 = C_1 \left[\frac{bS}{C_{os\Lambda_{1/4}}} \left(\frac{1+2\lambda}{3+3\lambda} \right) \left(\frac{M_{TOM}N}{S} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \quad (4.6.4)$$

Solving this equation for fore wing of Box 1 model gives a coefficient $C_1 = 0.028$. The values of C_1 for other fore wing models are shown in Table 4.3. The average is 0.028 and the variance is 2.51×10^{-8} . Notice the similarity of the weight coefficients. This is primarily because the values were rounded up to 3 decimal places. Furthermore, this similarity in values is indicative of the fidelity of the methodology and the linear correlation of the results.

The above process was also repeated for the aft wing and the coefficients obtained are in Table 4.4. Here as well the average coefficient is 0.028 and the variance 2.54×10^{-8} .

Thus, the empirical wing mass estimation equation for the fore and aft wings of a medium range box wing aircraft can be written as:

$$M_W = 0.028 \left[\frac{bS}{C_{os\Lambda_{1/4}}} \left(\frac{1+2\lambda}{3+3\lambda} \right) \left(\frac{M_{TOM}N}{S} \right)^{0.3} \left(\frac{V_D}{\tau} \right)^{0.5} \right]^{0.9} \quad (4.6.5)$$

Table 4.3: Fore Wing Models Coefficients

Model	Coefficient	Average	Variance
Box 1	0.028	0.028	2.51×10^{-8}
Box 2	0.028		
Box 3	0.028		
Box 4	0.028		
Box 5	0.028		
Box 6	0.028		
Box 7	0.028		
Box 8	0.028		
Box 9	0.028		
Box 10	0.028		

Table 4.4: Aft Wing Models Coefficients

Model	Coefficient	Average	Variance
Box 1	0.028	0.028	2.54×10^{-8}
Box 2	0.028		
Box 3	0.028		
Box 4	0.028		
Box 5	0.028		
Box 6	0.028		
Box 7	0.028		
Box 8	0.029		
Box 9	0.028		
Box 10	0.028		

4.7 Chapter Summary

This chapter outlined the challenges of predicting the wing mass of box wing aircraft and stressed the importance of wing mass estimation algorithms at an aircraft's conceptual design stage. After considering several equations, the C_1 method chosen for its adaptability and simplicity. Subsequently, a wing mass estimation correction coefficient was developed by relating conventional cantilever wings to box wing aircraft wings. The results indicate that the same correction coefficient would apply to both the fore and aft wings.

Wing/Tip Fin Joint Fixity

5.1 Introduction

In the design of a box wing aircraft an issue of consideration is the type of wing/tip fin joint fixity the aircraft should have. In an investigation by Lin et al²⁸ they show that a rigid joint is probably the best in reducing the wing root bending moment and is therefore, the optimum joint fixity for a classical joined wing aircraft configuration as sketched in Fig 5.1. However, they recommend caution when extrapolating their results to other joined-wing configurations. This chapter draws inspiration from their work and it is instructive to evaluate the type of joint fixity that would be more beneficial for the box wing configuration illustrated in Fig 5.2.

FEA were therefore performed to determine the best wing joint fixity of a statically loaded, idealized box wing configuration where both wings produce equal total lift forces. Thus, the internal stress resultants that arise from employing the 4 joint fixity types shown in Fig 5.3 for an assumed flight loading condition were analyzed.

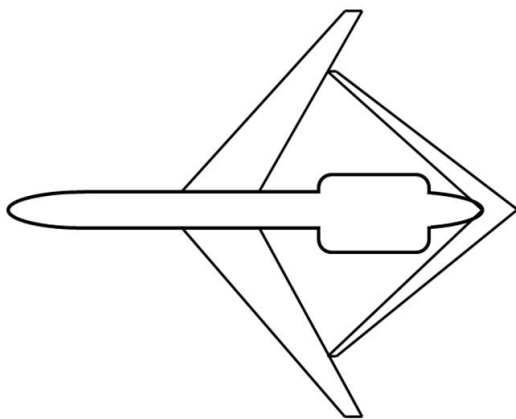


Figure 5.1: Joined Wing Configuration

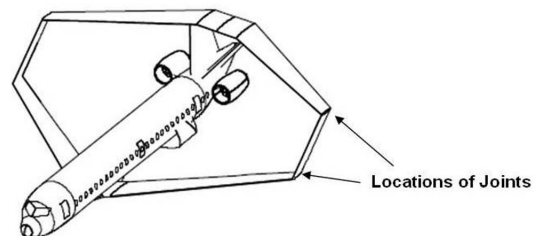


Figure 5.2: Box Wing Configuration

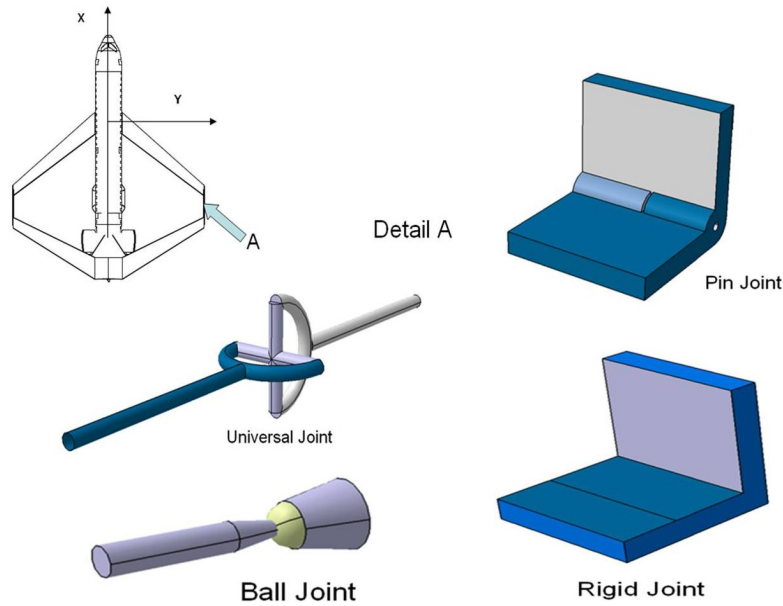


Figure 5.3: Joint Fixity Types

5.2 Aerodynamic Loads and Finite Element Analysis

The aerodynamic loads used to investigate the wing/tip fin joint fixity is as explained in Chapter 3 Section 5. Similarly, the finite element analysis and structural sizing procedure performed is as outlined in Chapter 3 Sections 6, 7 and 8.

5.3 Results

The comparative analysis of the moments and forces in the 4 wing tip joint configurations studied also includes the results of a cantilever wing which was evaluated along with the other joint types. It was basically the fore wing of the box wing configuration with the joint disconnected and provides comparison with the trends of the box wing.

Trim load was estimated and added to the overall mass of the aircraft before running the simulation. AVL⁵⁵ was used to shape the lift distribution in order to match the lift and trim together. The parameters evaluated were torque, out-of-plane shear force, out-of-plane bending moment, drag-wise bending moment and wing tip deflection.

Figures 5.4 and 5.5 show the torsional load trends in the fore and aft wings for each joint fixity. The applied torsional loads were taken at the elastic axis and moments about the elastic axis of one wing were also caused by the applied loads on the other wing transmitted through the wing joint. The exception is the cantilever wing and Fig 5.4 shows that the torsional moment falls to zero towards the wing tip in this case. However, of the four joint fixities, the rigid joint produced the largest torsional moment. This is because unlike the universal, pin and ball joints, the rigid joint transmits all moments

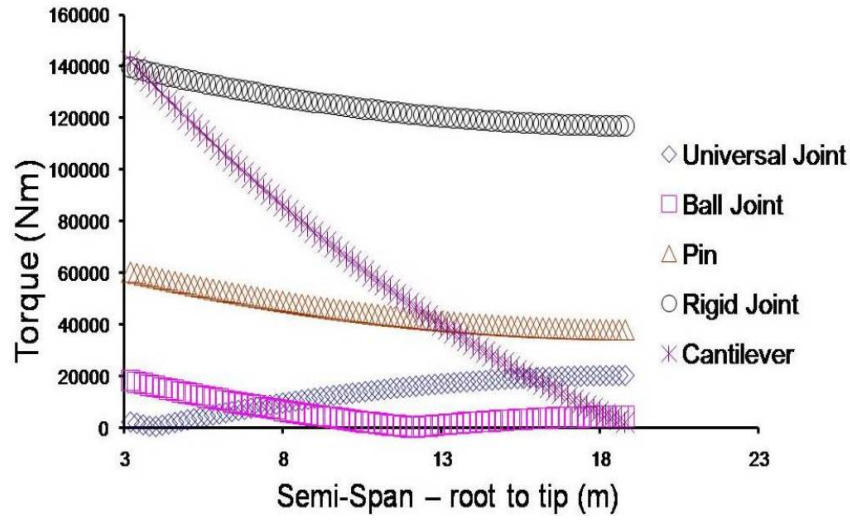


Figure 5.4: Fore Wing Torsional Force Distribution

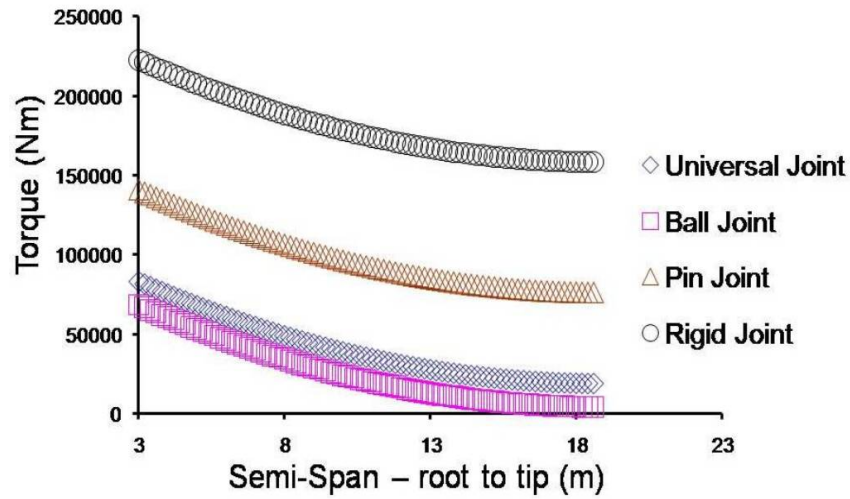


Figure 5.5: Aft Wing Torsional Force Distribution

and forces through the wing joint.

Figure 5.6 and 5.7 present the out-of-plane shear force for both fore and aft wings. The trends for all the joints and the cantilever wing are similar. However, for the box wing configuration, the rigid joint produces marginally lower out-of-plane shear stresses than all the other joints on both wings including the cantilever wing.

Figure 5.8 and 5.9 represent the out-of-plane bending moment distributions in the fore and aft wings. The trends for the fore and aft wings for all joint types are similar as far as out-of-plane bending moments are concerned. All the joint types show lower wing root bending moment than the cantilever wing. However, the rigid joint shows the greatest reduction of out-of-plane bending moment stresses at the wing root.

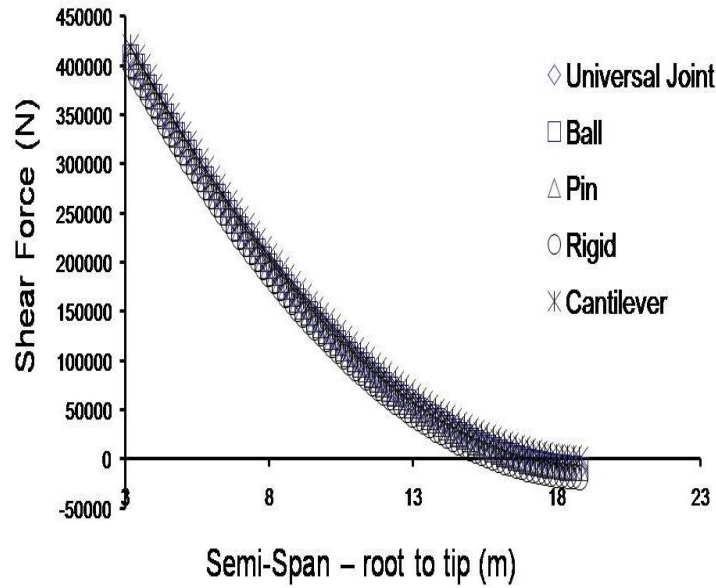


Figure 5.6: Fore Wing Out-of-Plane Shear Force Distribution

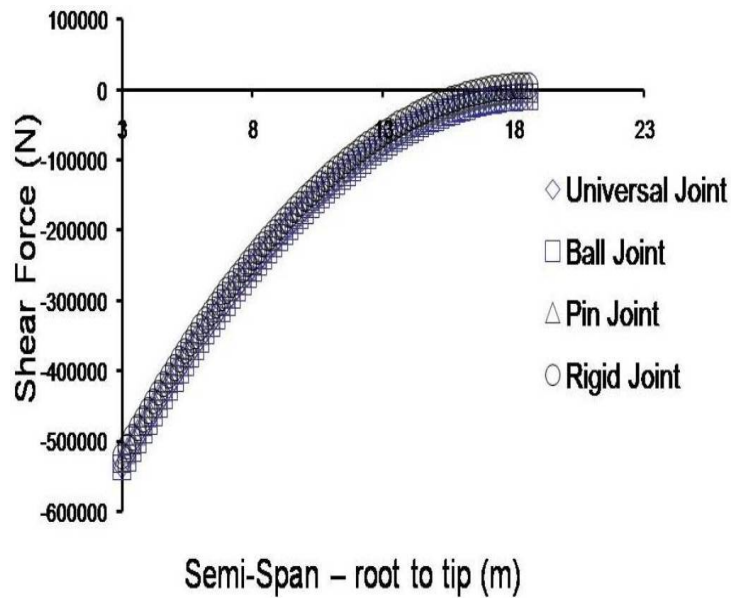


Figure 5.7: Aft Wing Out-of-Plane Shear Force Distribution

Furthermore, the relatively high out-of-plane bending moments produced by the other 3 joints could make the aft wing susceptible to lateral buckling or divergence and have possibly lower flutter speeds. Therefore, a non-linear analysis is recommended for future work to identify post buckling behaviour of the system.

The drag-wise bending moment distributions in the fore and aft wings are represented in Fig 5.10 and 5.11. It should be noted that the drag-wise bending moment is usually not

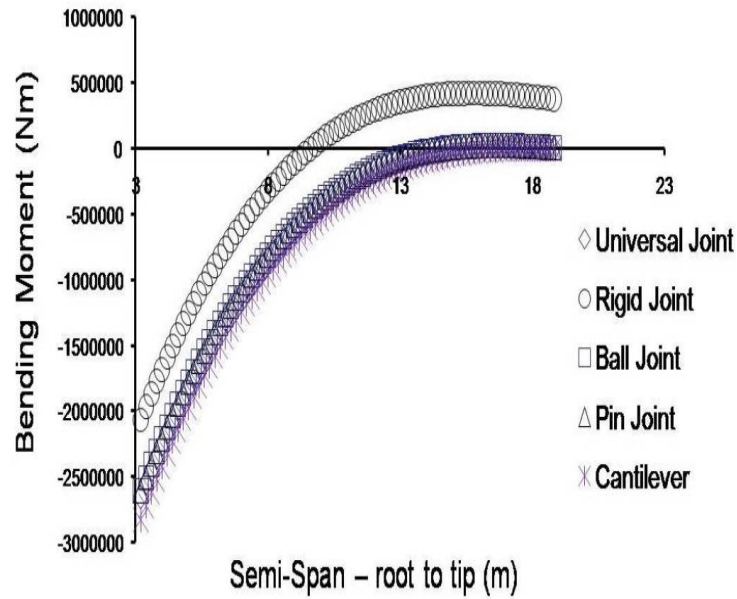


Figure 5.8: Fore Wing Out-of-Plane Bending Moment Distribution

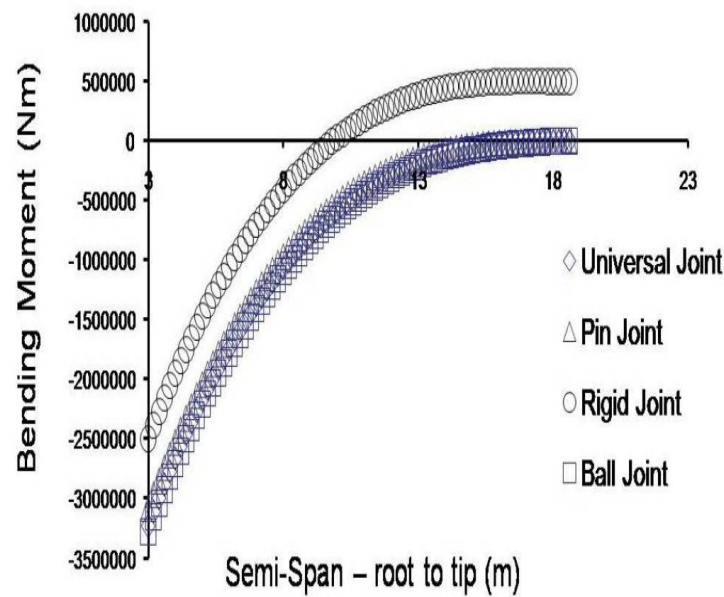


Figure 5.9: Aft Wing Out-of-Plane Bending Moment Distribution

of significance in a cantilever wing as Fig 5.10 highlights. Its prominence in the box wing has to do with the inclined tip fin and sweep of the wings (see Figs 5.2 and 4.6(a)).

Under load, the wing tips of the fore and aft wings tend to displace upwards and away from each other. This increases torque moments on the wings as well as generating drag-wise bending moments on both wings. The drag-wise bending moment could therefore

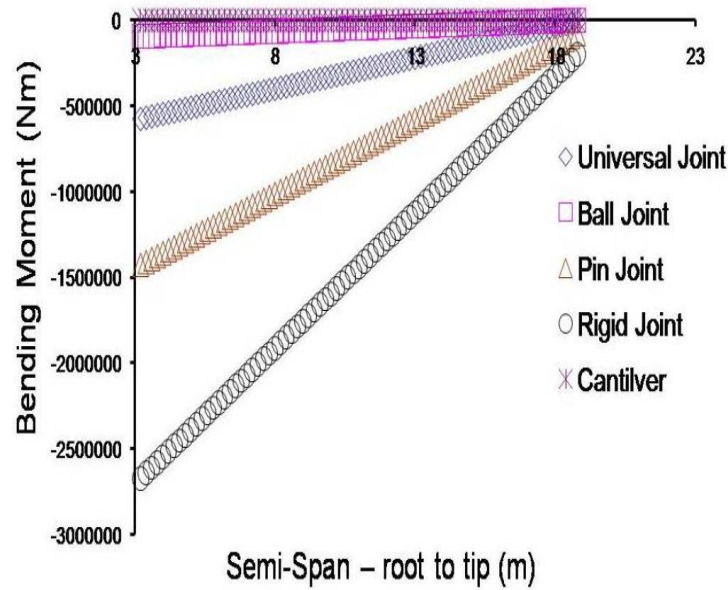


Figure 5.10: Fore Wing Drag-wise Bending Moment Distribution

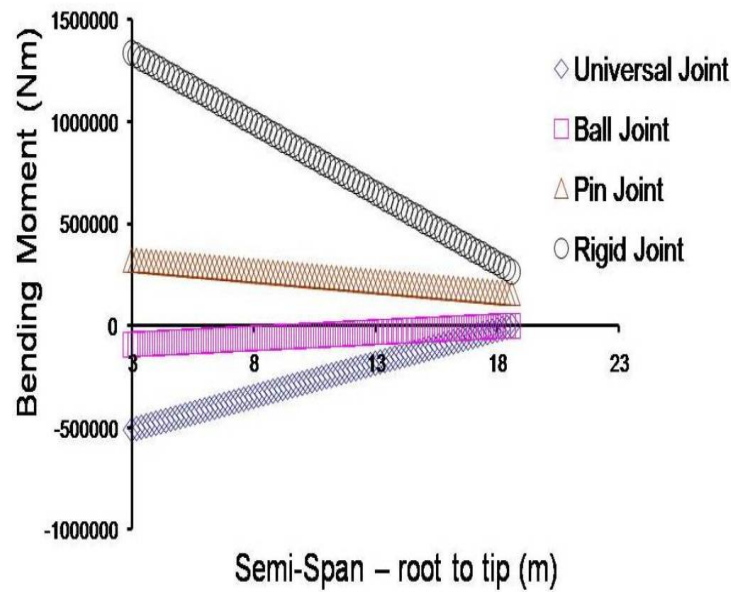
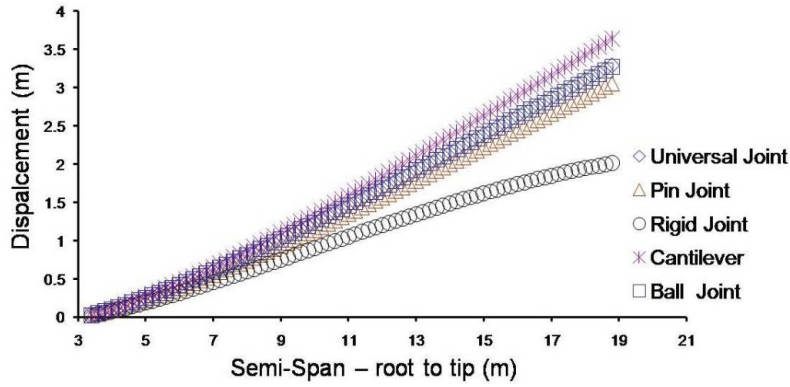
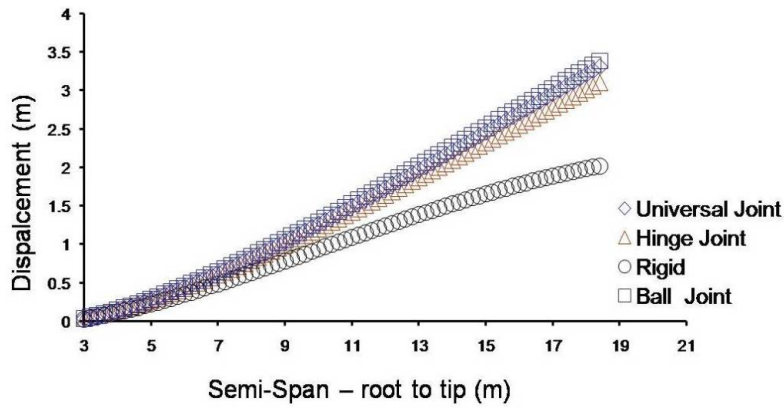


Figure 5.11: Aft Wing Drag-wise Bending Moment Distribution

be considered in sizing the wing torsion box of box wing aircraft. Furthermore, unlike the 3 other joints, the rigid joint transmits all the drag-wise bending moment and thus is significantly higher than all the other joint types. Also noteworthy is that in the fore wing the drag-wise bending moments are significantly higher than at the aft wing and show a steep linear decrease towards the wing tip. At the aft wing the decrease is shallower and this difference could be attributable to the lower sweep angle of the aft wing.



(a) Fore Wing Deflection



(b) Aft Wing Deflection

Figure 5.12: Wing Deflections

Figure 5.12 represents the wing tip deflections for the joint fixities and the cantilever wing. As expected the cantilever wing deflects the most, at almost twice that of the rigid joint. This figure highlights the rigidity of box wing configuration with the rigid joint being the most rigid.

Tables 5.1 and 5.2 summarize the results of the relative stress levels for the fore and aft wings. The wing root values of the joints have been normalized with respect to the respective rigid joint value. For the fore wing all the other joints transmit at least about 26% more out-of-plane bending moment stresses to the wing root than the rigid joint while the cantilever wing transmits 37% more. The ball and universal joints drag-wise bending moments are significantly less than that of the rigid joint at the wing root while the pin joint is about half that of the rigid joint. The trend of the out-of-plane shear force is that all joint types including the cantilever wing are marginally higher than that of the rigid joint. For torsional forces, the rigid joint is similar to the cantilever wing and transmits much higher stresses to the wing root than any of the other joints, with the universal joint transmitting the least amount. As for wing tip deflections, the cantilever wing deflects 81% more than that of the rigid joint box wing configuration. Overall, the universal, ball and pin joints deflect at least 50% more than the rigid joint.

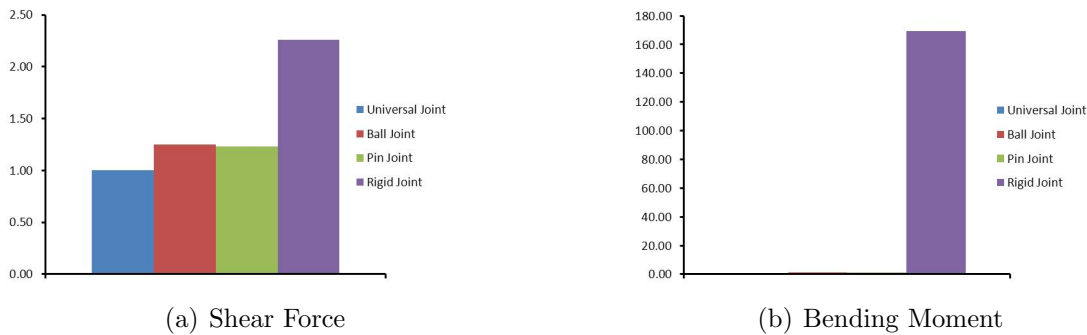
Table 5.1: Normalized Wing Root Parameters - Fore Wing

Output	Rigid	Universal	Ball	Pin	Cantilever Wing
Out-of-plane BM	1	1.28	1.26	1.26	1.37
Dragwise BM	1	0.21	0.04	0.53	0.00
Out-of-plane SF	1	1.03	1.02	1.02	1.05
Torsion	1	0.02	0.13	0.43	1.02
Tip Deflection	1	1.64	1.62	1.52	1.81

Table 5.2: Normalized Wing Root Parameters - Aft Wing

Output	Rigid	Universal	Ball	Pin
Out-of-plane BM	1	1.29	1.31	1.23
Dragwise BM	1	0.39	0.07	0.24
Out-of-plane SF	1	1.03	1.03	1.01
Torsion	1	0.37	0.31	0.63
Tip Deflection	1	1.64	1.68	1.54

The aft wing behaves in a similar manner to the fore wing in that all the other joints transmit more than 23% out-of-plane bending moment stresses to the wing root than the rigid joint. Again, the trend for drag-wise bending moment is opposite to the out-of-plane bending moment. The pin and universal joints transmit 7% and 24% less than that of the rigid joint while the universal joint is about 40% the value of the rigid joint. The trends of the out-of-plane shear force show that all the other joints transmit marginally more shear force to the wing root than the rigid joint. All the other joints transmit lower torsional stresses to the wing root than the rigid joint because the rigid joint transmits all the moments and forces. For the wing tip deflection, the trend is similar to the fore wing with all joints deflecting at least 50% more the the rigid joint.

**Figure 5.13:** Normalized Fore Wing/Tip Fin Joint Stresses

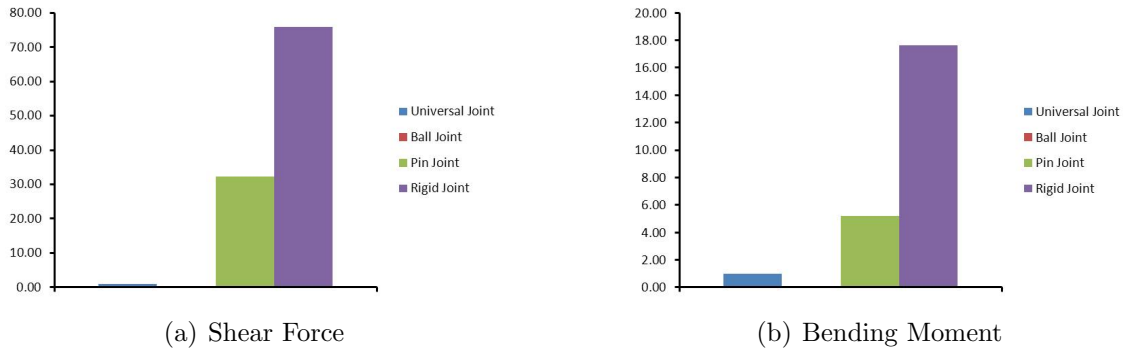


Figure 5.14: Normalized Aft Wing/Tip Fin Joint Stresses

Figure 5.13 shows the shear force and bending moment in the fore wing/tip fin normalized with respect to the universal joint value. In Fig 5.13(a) the ball and pin joint transmit about 25% more shear stresses than the universal joint. For the rigid joint it transmits over 200% more shear stress compared to the universal joint. Figure 5.13(b) shows the bending moments for the 4 joints all normalized with respect to their universal joint value. The rigid joint transmit over 16000% more bending moments than all the other joints.

Figure 5.14 shows the shear force and bending moment in the aft wing/tip fin normalized with respect to the universal joint value. In Fig 5.14(a) the shear stresses transmitted by the universal and ball joints are negligible compared to the pin and rig joints. For the rigid joint it transmits over 7500% more shear stress compared to the universal joint. Figure 5.14(b) shows the bending moments for the 4 joints all normalized with respect to their universal joint value. The rigid joint is dominant and transmits over 1200% more bending moments than all the other joints.

5.4 Chapter Summary

It has been established by Lin et al²⁸ that a rigid joint is the optimum joint fixity for a classical joined wing aircraft configuration. This chapter has examined the structural issue of the wing/tip fin joint fixity for box wing aircraft. The universal, pin, ball and rigid joint fixity types were analyzed by using loads from a vortex lattice tool to perform finite element analysis. The results indicate that the rigid joint would be the preferred joint fixity.

Tip Fin Inclination Effect on Wing Design

6.1 Introduction

Box wing aircraft wings are joined by a tip fin but what should be its position relative to the aircraft's vertical axis. Whether it is vertical or inclined would influence the aircraft's wing positioning, mass and cg issues and stability and control. This chapter examines the effect of tip fin inclination on box wing aircraft design. With regards to box wing overall wing relative positioning, Bagwill and Selberg¹⁹ posit that a positively staggered wing arrangement is more efficient than the negative; by positive stagger they refer to the higher wing being in front of a lower aft wing and by negative stagger they refer to the reversed arrangement. However, Prandtl⁸ highlights the beneficial influence of a maximized vertical separation for biplane configurations. Furthermore, work by Smith and Jemitola⁶⁰ showed that the negatively staggered arrangement benefits from the presence of the tail fin as a natural way of maximizing wing vertical separation; the same is not achievable on the positively staggered arrangement without significant mass penalties and directional stability compensation issues.

The structural effects of tip fin inclination on the design of a negatively staggered medium range box wing airliner are unclear. Studies were therefore performed to evaluate the structural consequences of changing the tip fin inclinations in the wing assembly structure for an assumed flight loading condition. Changing the tip fin inclination, Fig 6.1, requires either displacing the wings longitudinally relative to each other or stretching the fuselage and for this analysis the former was implemented to change tip fin inclination.

6.2 Aerodynamic Loads and Finite Element Analysis

The aerodynamic loads used to investigate the structural effects of tip fin inclination is as explained in Chapter 3 Section 5. Similarly, the finite element analysis and structural sizing procedure performed is as outlined in Chapter 3 Sections 6, 7 and 8.

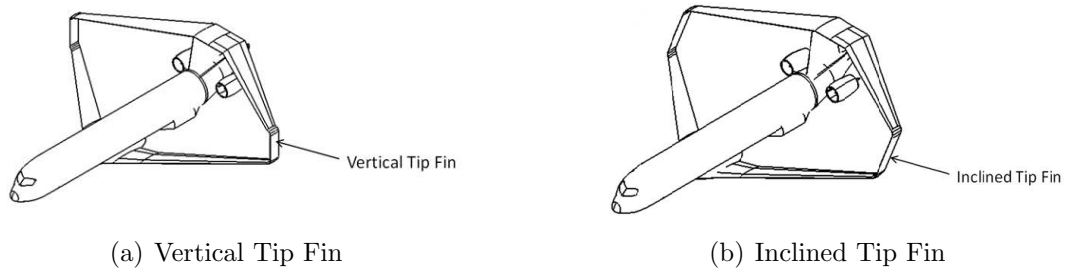


Figure 6.1: Tip Fin Inclinations

6.3 Results

Figure 6.2 shows the torsional load trends in the fore and aft wings for each tip fin inclination. The applied torsional loads were taken at the elastic axis and moments about the elastic axis of one wing were also caused by the applied loads on the other wing transmitted through the wing joints. Fig 6.2 shows that the torsional moment in the fore wing reduces and becomes increasingly negative as the tip fin inclination increases. The same is true for the torsional distribution within the tip fin although the changes are marginal. The figure shows that at the fore wing root there is a 90% decrease in torsional loads from the vertical tip fin to the 40° inclined tip fin. In the aft wing the pattern is opposite what obtains in the fore wing. Here the change from the vertical to the 40° inclined tip fin is an increase of 28% at the wing root.

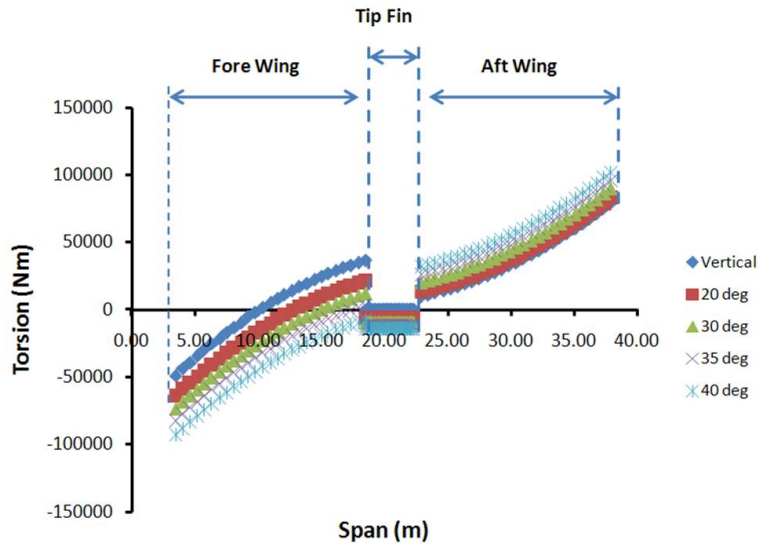


Figure 6.2: Torsional Force Distribution

The overall torsional distribution is affected by the behaviour of the wing tips which twist upwards and away from each other under load as illustrated in Fig 6.3; the fore wing tip twists nose down and the aft wing tip twists nose up. Thus, the fore wing is subjected

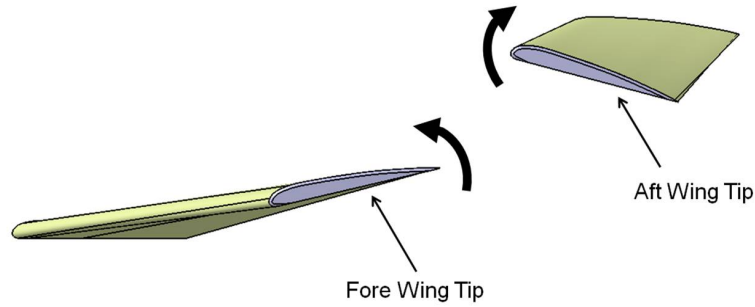


Figure 6.3: Wing Tips Displacement under Load

to additional torsional strain apart from that due to airfoil pitching moment while at the aft wing there is an alleviating effect on the wing's airfoil pitching moment. These, in addition to sweep angle mismatch accounts for the significant differences in torsional force distribution of the fore and aft wings shown in Fig 6.2. The torsional forces put extra strain on the tip fin in the form of in-plane shear forces which is not a very visible effect because what drives the sizing is out-of-plane stresses. However, it would be expected that these torsional forces would affect the design of the tip fin/wing joints.

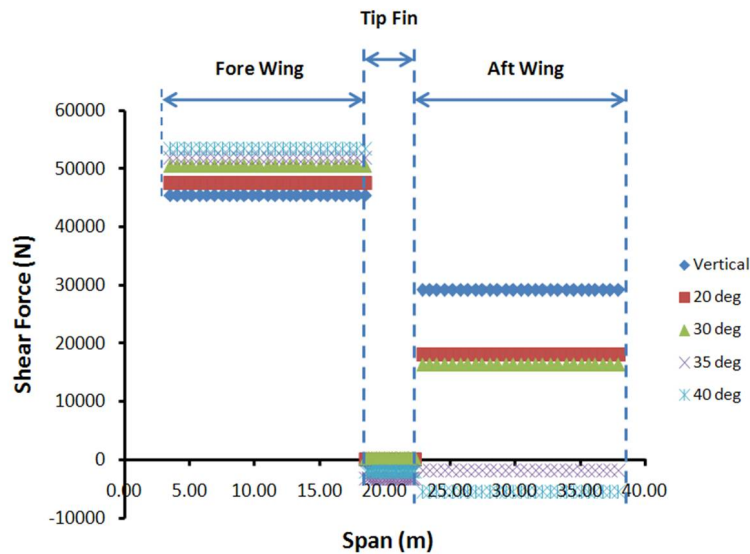


Figure 6.4: Dragwise Shear Force Distribution

The drag wise shear force is not usually a factor in sizing load bearing members in conventional wings. However, it is instructive to note the trend in a box wing aircraft as shown in Fig 6.4. In the fore wing the wing root shear force shows an increase with tip fin inclination with the 40° inclined tip fin being 17% greater than that of the vertical tip fin. In the tip fin itself the trend is reversed with the 40° inclined tip fin being 30% less than that of the vertical. The trend in the aft wing shows a more amplified distribution of shear force as a function of tip fin inclination. The overall trend in the aft wing is

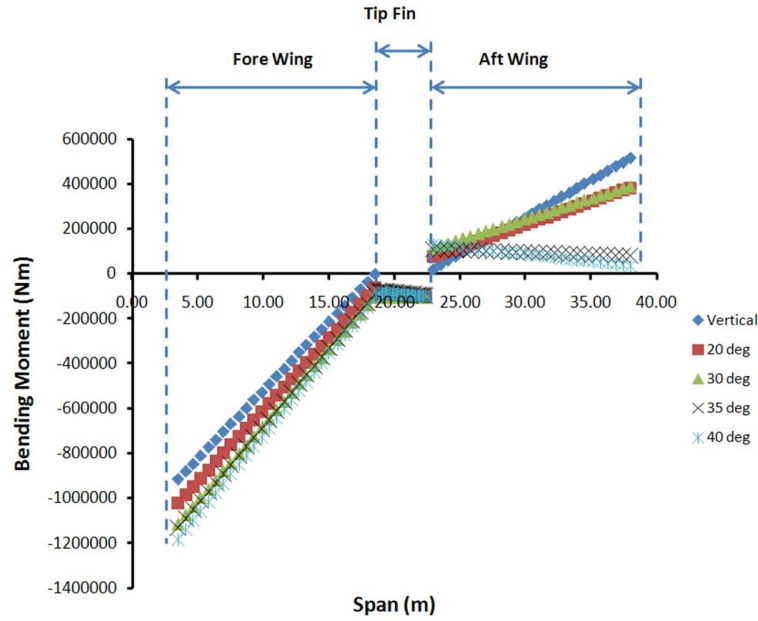


Figure 6.5: Dragwise Bending Moment Distribution

reversed with respect to the fore wing. The aft wing root shear force in the case of 40° inclined tip fin is 81% smaller than the same force in the case of the vertical tip fin

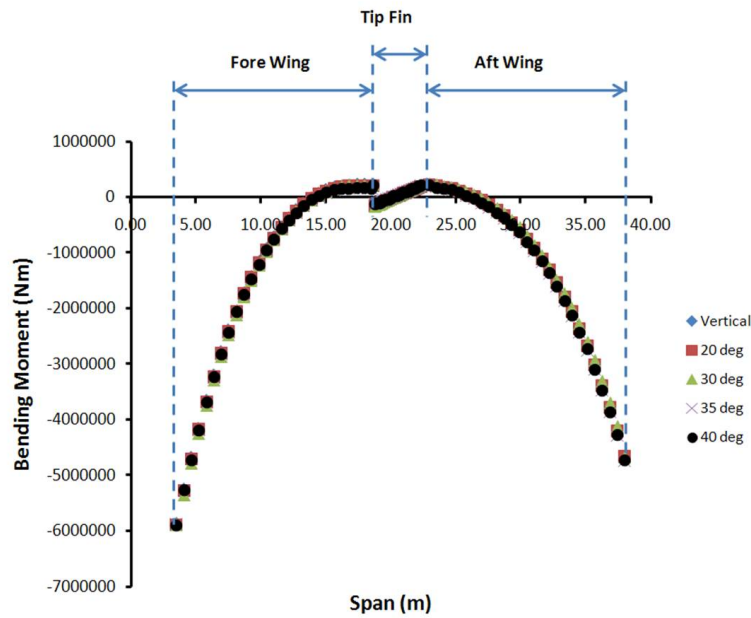


Figure 6.6: Bending Moment Distribution

Fig 6.5 shows the dragwise bending moment distribution. At the fore wing the wing root dragwise bending moment shows an increase in magnitude of 29% from the vertical tip fin to the 40° inclined tip fin. At the tip fin itself the changes are marginal while at the

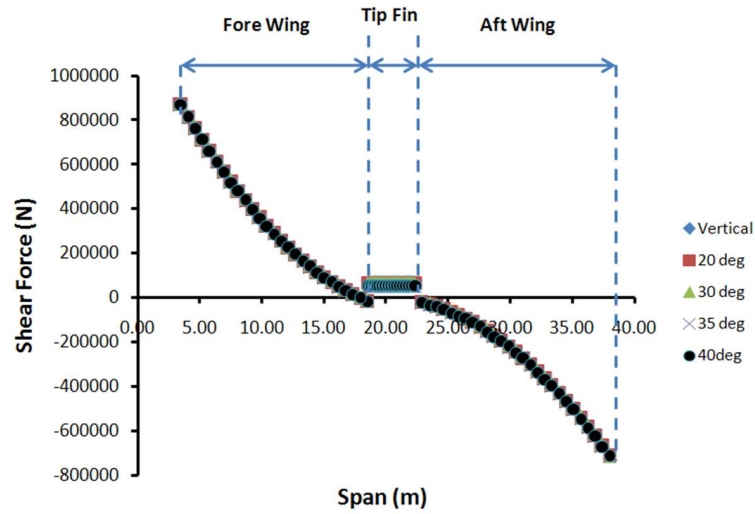


Figure 6.7: Shear Force Distribution

aft wing the trend is more dispersed than in the fore wing. Here the wing root dragwise bending moment of the 40° inclined tip fin is just 7% of the value of the vertical tip fin.

Table 6.1: Normalized Wing Root Parameters - Fore Wing

Tip Fin Inclination	Vertical	20 deg	30 deg	35 deg	40 deg
Bending Moment	1.00	1.01	1.01	1.01	1.01
Shear Force	1.00	1.00	1.00	1.00	1.00

Table 6.2: Normalized Wing Root Parameters - Aft Wing

Tip Fin Inclination	Vertical	20 deg	30 deg	35 deg	40 deg
Bending Moment	1.00	1.00	1.01	1.02	1.02
Shear Force	1.00	1.00	1.01	1.01	1.01

The out-of-plane bending moment distributions are essentially the same in the fore wing, tip fin and aft wing for all tip fin inclinations as shown in Fig 6.6. The same can be said of the out-of-plane shear force distribution shown in Fig 6.7. The little difference between the tip fin inclination models is also shown in Tables 6.1 and 6.2. Here the fore and aft wing root values of the five tip fin inclinations were normalized with respect to the respective vertical tip fin value and the results show little differences.

The wing deflections in Fig 6.8 show little variation with tip fin inclination and is in harmony with the out-of-plane bending moment distribution shown in Fig 6.6. Table 6.3 is the result of normalizing the wing tip deflections, tip fin torsion box masses and overall wing torsion box masses by their respective vertical tip fin values. Table 6.3 shows that the vertical tip fin is stiffer than the others by a maximum of 6%. Although, the 40° inclined tip fin is 23% heavier than the vertical tip fin by way of torsion box mass the overall

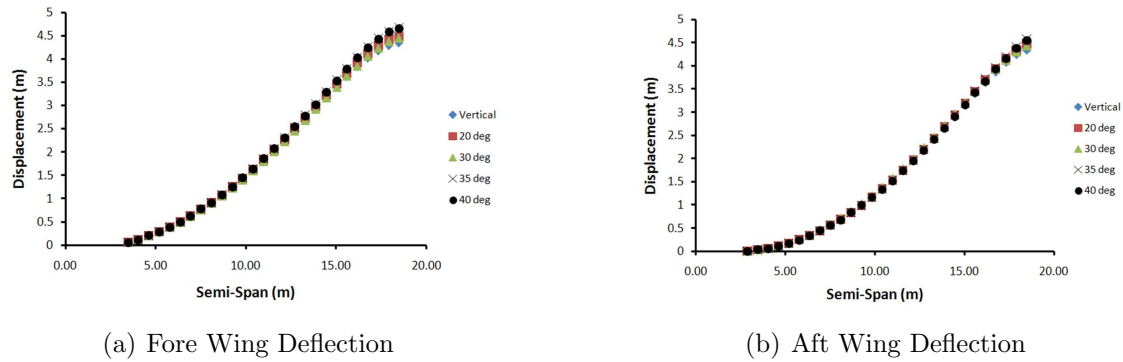


Figure 6.8: Wing Deflections

Table 6.3: Normalized Wing Parameters

Tip Fin Inclination	Vertical	20 deg	30 deg	35 deg	40 deg
Wing Tip Deflection	1.00	1.03	1.05	1.06	1.06
Tip Fin Torsion Box Mass	1.00	1.00	1.01	1.04	1.23
Wing Torsion Box Mass	1.00	1.00	1.00	1.00	1.02

effect is a maximum of 2% in difference between the configuration wing torsion box masses.

6.4 Chapter Summary

This chapter has presented an examination of the effect of tip fin inclination of the structural design of the box wing aircraft. It has established that a negatively staggered box wing benefits from the presence of the tail fin to maximize wing vertical separation. Five tip fin inclinations, 0, 20, 30, and 40 degrees were examined by using loads from a vortex lattice tool to perform finite element analysis. The results indicate that tip fin inclination may have minimal structural effect on box wing aircraft design.

Case Studies

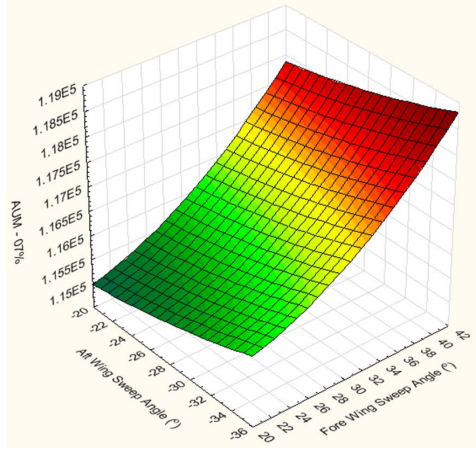
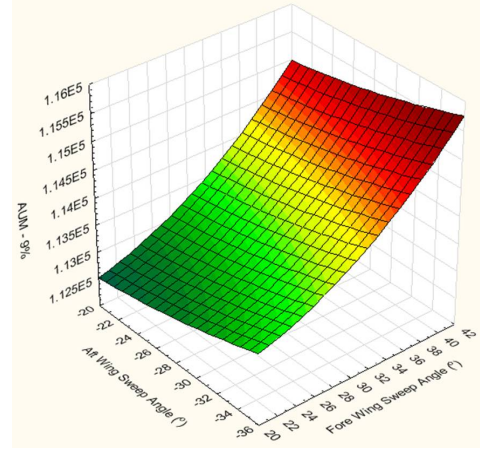
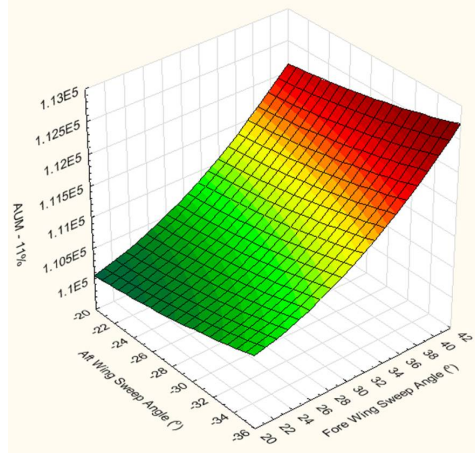
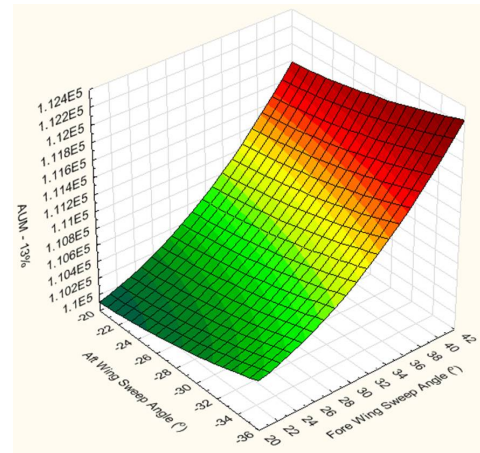
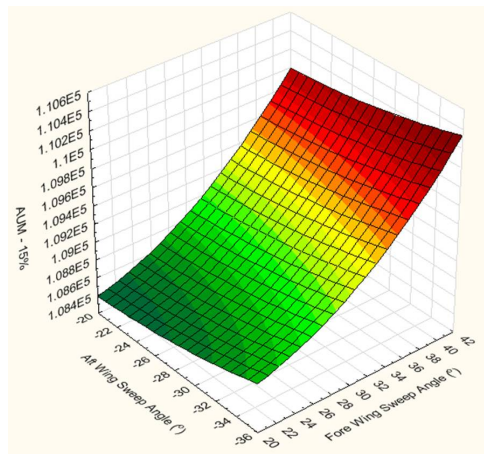
This chapter describes the use of the optimization tool to perform optimizations on thickness to chord ratio and wing area. The optimizations were performed as functions of other key aircraft parameters using response surfaces and regression analysis.

7.1 Average Thickness to Chord Ratio Optimization

7.1.1 τ Optimization - Box Wing Aircraft

The average thickness to chord ratio, τ , should be as large as possible for minimization of wing structural weight and to provide volumetric capacity for fuel. On the other hand τ needs to be as small as possible for drag reduction reasons. Thus, an optimization procedure was carried out to select the optimum τ for the box wing using the baseline box wing aircraft parameters. The first phase involved studying τ as a function of AUM for τ values of 7, 9, 11, 13 and 15 percents. Here the variable was wing sweep angle and the objective function minimization of AUM. A total of 1235 models were created and by using the VBA macros called RangeGSeek and SelectAUM each model was optimized. The results were then used to create a response surfaces for each τ as shown in Fig 7.1. The response surfaces show that higher values of τ could be suitable as they make for lower AUMs. There is a 6% reduction in AUM from $\tau = 7\%$ models to $\tau = 15\%$ models. Fig 7.1 also shows that lower sweep angles for fore and aft wings would produce an overall lighter aircraft. These trends are consistent with literature as regards the significance of wing sweep angle on aircraft AUM. The analysis would therefore suggest that the model with fore and aft wing sweep angles of 20 and -20 degrees respectively and $\tau = 15\%$ would be optimum.

However, to provide a different perspective another optimization procedure was performed. In this second study τ as a function of drag, C_D , was evaluated for τ values of 7, 9, 11, 13 and 15 percents. As in the preceding study the variable was wing sweep angles but the objective function was minimization of C_D . Another 1235 models were created and by using the VBA macro called SelectCDo each model was optimized. The results were then used to create response surfaces as shown in Fig 7.2. The response surfaces show trends opposite to that of the previous study. For each response surface

(a) AUM - $\tau = 0.07$ (b) AUM - $\tau = 0.09$ (c) AUM - $\tau = 0.11$ (d) AUM - $\tau = 0.13$ (e) AUM - $\tau = 0.15$ **Figure 7.1:** Response Surfaces for AUM as a function of τ

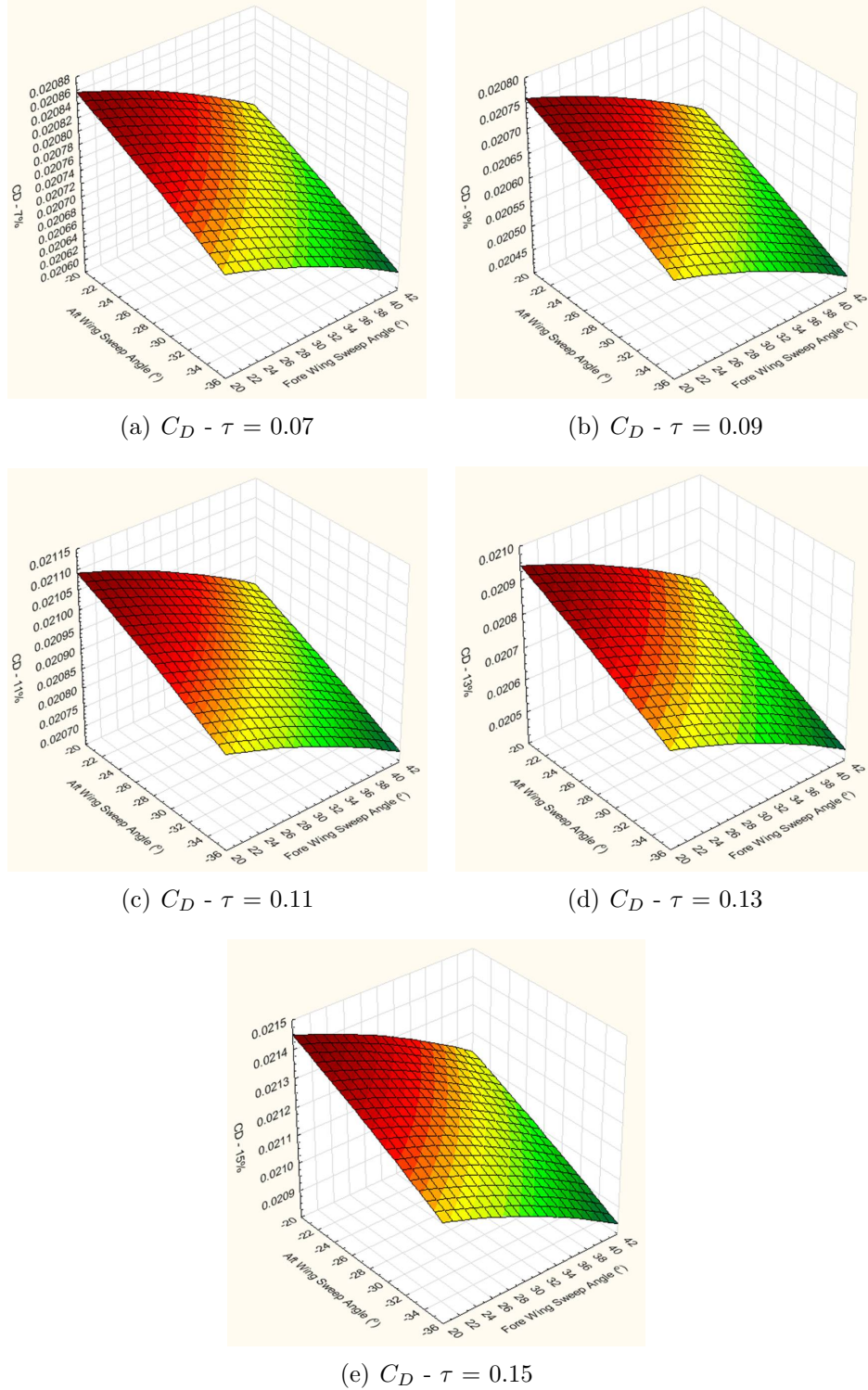
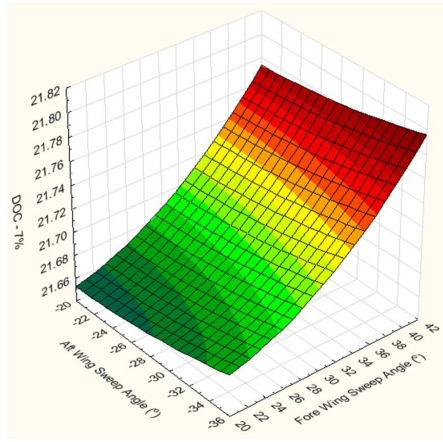
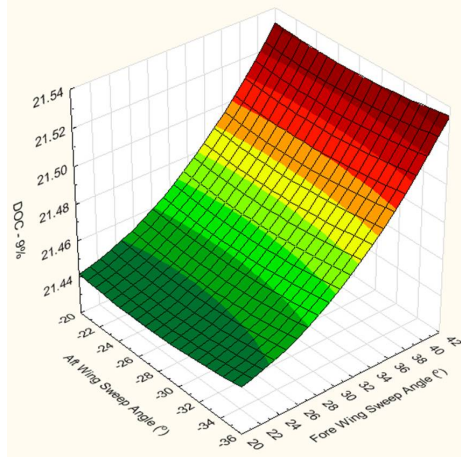
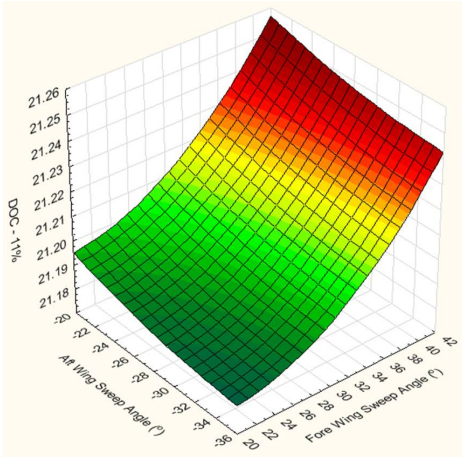
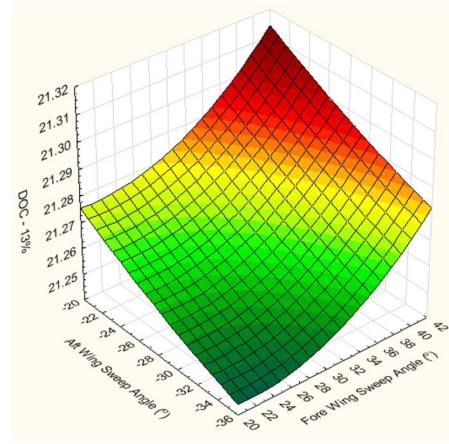
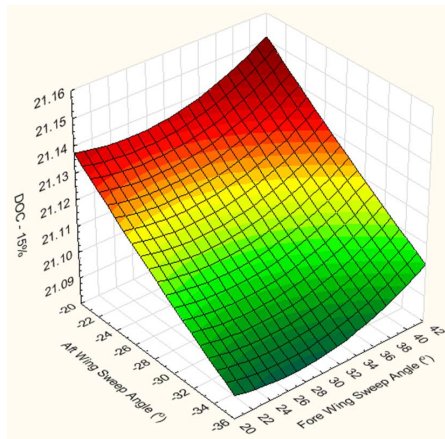


Figure 7.2: Response Surfaces for C_D as a function of τ

C_D increases with reduction in wing sweep angles. Furthermore, C_D also increases with increase in τ . There is a 2% increase in C_D from $\tau = 7\%$ models to $\tau = 15\%$ models. This

(a) DOC - $\tau = 0.07$ (b) DOC - $\tau = 0.09$ (c) DOC - $\tau = 0.11$ (d) DOC - $\tau = 0.13$ (e) DOC - $\tau = 0.15$ **Figure 7.3:** Response Surfaces for DOC as a function of τ

analysis suggests the model with fore and aft wing sweep angles of 42 and -36 degrees respectively and $\tau = 7\%$ would be optimum. This optimum is at the opposite extreme of the optimum in the preceding optimization.

Hence, a third optimization procedure was performed still studying τ values of 7, 9, 11, 13 and 15 percents. As in the preceding study the variable was wing sweep angles but the objective function minimization of *DOC*. Another 1235 models are created and by using the VBA macro called AcCost each model was optimized. The results were then used to create response surfaces as shown in Fig 7.3. Unlike the preceding optimizations each response surface shows a different minimum but overall high sweep angles means high *DOC*. For $\tau = 7\%$ the minimum is a model with fore and aft sweep angles of 20 and -20 degrees respectively. For $\tau = 9\%$ the minimum is a model with fore and aft sweep angles of 20 and -28 degrees respectively. For $\tau = 11\%$ the minimum shifts to the model with fore and aft sweep angles of 20 and -36 degrees respectively. At the $\tau = 13\%$ response surface the minimum is a model with fore and aft sweep angles of 22 and -36 degrees respectively. Finally for $\tau = 15\%$ the minimum is a model with fore and aft sweep angles of 27 and -36 degrees respectively. There is a 3% reduction in *DOC* from $\tau = 7\%$ models to $\tau = 15\%$ models. This analysis suggests the model with fore and aft wing sweep angles of 27 and -36 degrees respectively and $\tau = 15\%$ would be optimum.

From the 3 optimizations performed and response surfaces in Figs 7.1, 7.2 and 7.3 the conflict in the desirable characteristics is evident and this is typical of aircraft optimization. An optimum τ was therefore to be determined but considering the entire 3705 models. Hence, the outcomes of the 3 previous optimizations were normalized using their respective minimum values. By adding up the normalized values of C_D , AUM and *DOC* for each τ a function of suitability, gamma (γ), was created as shown in the equations below.

$$(\gamma)_{\tau} = \left(\frac{C_D}{(C_D)_{min}} \right) + \left(\frac{AUM}{(AUM)_{min}} \right) + \left(\frac{DOC}{(DOC)_{min}} \right) \quad (7.1.1)$$

where

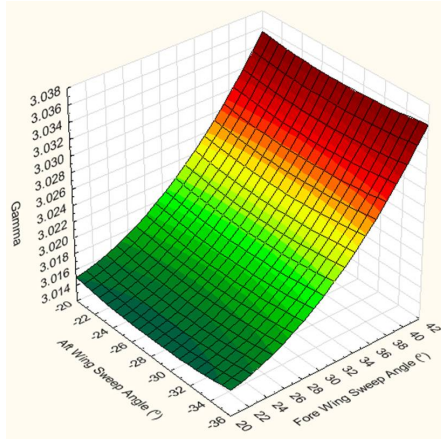
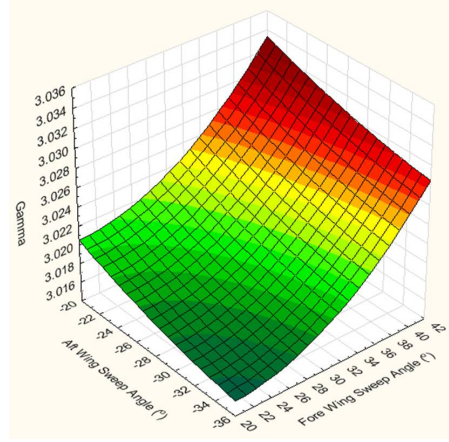
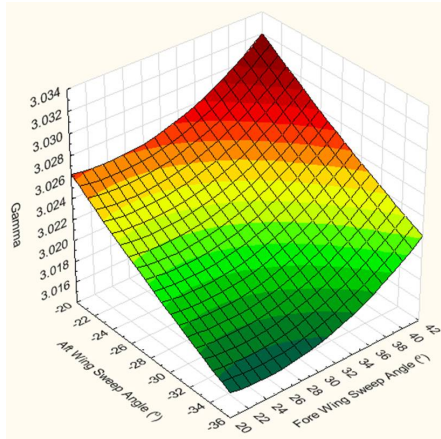
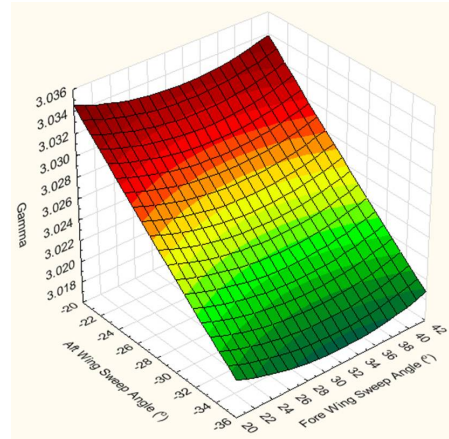
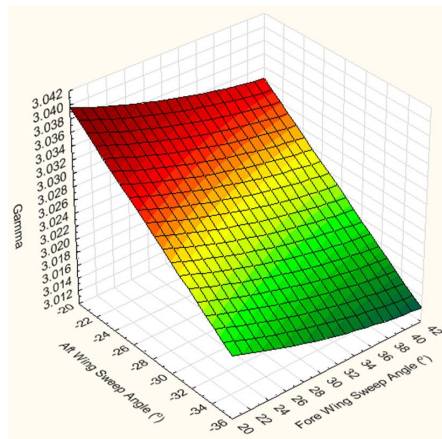
$$C_D = C_D(\mathbf{\Lambda}_F, \mathbf{\Lambda}_A, \tau),$$

$$AUM = AUM(\mathbf{\Lambda}_F, \mathbf{\Lambda}_A, \tau)$$

and

$$DOC = DOC(\mathbf{\Lambda}_F, \mathbf{\Lambda}_A, \tau)$$

Note that all factors in Eqn 7.1.1 affect each other directly and indirectly; this is typical of the intricately interdependent process of aircraft design. As the computation was performed simultaneously, there was no duplication or layering of the process. The benefit of Eqn 7.1.1 is that the best of all the variables as a function of average thickness to chord ratio can be represented by one function and enables the decision making process with regards to the optimum value for τ .

(a) Response Surface for $(\gamma) - \tau = 07\%$ (b) Response Surface for $(\gamma) - \tau = 09\%$ (c) Response Surface for $(\gamma) - \tau = 11\%$ (d) Response Surface for $(\gamma) - \tau = 13\%$ (e) Response Surface for $(\gamma) - \tau = 15\%$ **Figure 7.4:** Response Surfaces for various τ

γ was used to create response surfaces for each τ as shown in Fig 7.4. In this optimization procedure the objective functions were the minimization of C_D , AUM and DOC. Thus,

the lower the value of gamma the better the model. The response surface for $\tau = 7\%$, Fig 7.4 (a), shows that fore and aft wing sweeps of 20 and -28 degrees respectively is the optimum. In Fig 7.4 (b) for $\tau = 9\%$ the optimum is the model with fore and aft wing sweeps of 20 and -36 degrees respectively. In Fig 7.4 (c) for $\tau = 11\%$ the optimum shifts to fore and aft sweep angles of 24 and -36 degrees respectively. For $\tau = 13\%$, Fig 7.4 (d), the optimum model is one with fore and aft wing sweeps of 32 and -36 degrees respectively. For Fig 7.4 (e) where $\tau = 15\%$ the optimum is the model with sweep angles of 42 and -36 degrees for the fore and aft wings respectively.

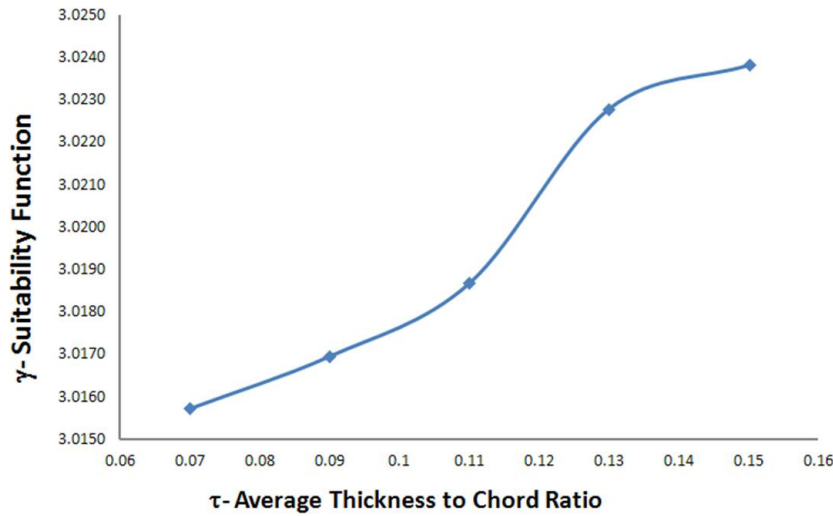


Figure 7.5: γ against τ

To narrow down the selection wing sweep angles were considered. Generally, wing sweep helps to delay the onset of the adverse effects of compressibility, delay the attainment of drag divergence mach number and push critical mach numbers to higher values. However, it is necessary for the angles to be as low as possible because as sweep angle increases so does induced drag and wing mass. In addition, there is a decrease in the maximum lift coefficient of the wing⁷⁶ with increase in sweep angle. Furthermore, high lift devices like trailing edges flaps perform poorly on swept wings. For forward swept wings there is the added issue of aeroelastic divergence with increase in sweep angle⁵⁴.

Thus, aft wing sweep angles below -25 degrees were eliminated and fore wing sweep angles below 29 and above 35 were removed. Using the chart (based on statistical methods) in Jenkinson⁵⁴ fore and aft wing sweep angles of 30 and -22 degrees were chosen as the minimum. The plotting of the minima of γ for this model and for all the other models response surface against τ gives Fig 7.5. This trend in this graph is consistent for all combinations of fore and aft sweep angles with respect to τ . The graph indicates that $\tau = 7\%$ model is the optimum however by checking if wing fuel volume satisfies the design range τ values below 9% do not satisfy this requirement. Thus the optimum τ for this box wing aircraft is 9%.

7.1.2 τ Optimization - Conventional Cantilever Aircraft

The preceding optimization procedure was also performed for the baseline conventional cantilever configuration. The sweep angles considered were from 22 to 40 degrees and as in the preceding analysis the objective functions were minimization of C_D , AUM and DOC . A total of 380 models of the conventional aircraft were made in the process. The plot of minimum γ against τ is shown in Fig 7.6. The general trend in the plots is that as sweep angle increases so does γ . It also shows some convergence for all sweep angles at $\tau = 15\%$. Due to the effects of drag divergence mach number sweep angles less than 30 degrees were eliminated in the selection process. This leaves the 30 degree wing sweep as the optimum plot in Fig 7.6 and its minimum is at $\tau = 11\%$. Thus, a τ value of 11% is the optimum for this baseline conventional cantilever aircraft. This value close to the equivalent real life aircraft of this study; the Boeing 767-200. It has a τ value of 11.5% which provides validation to the optimization performed.

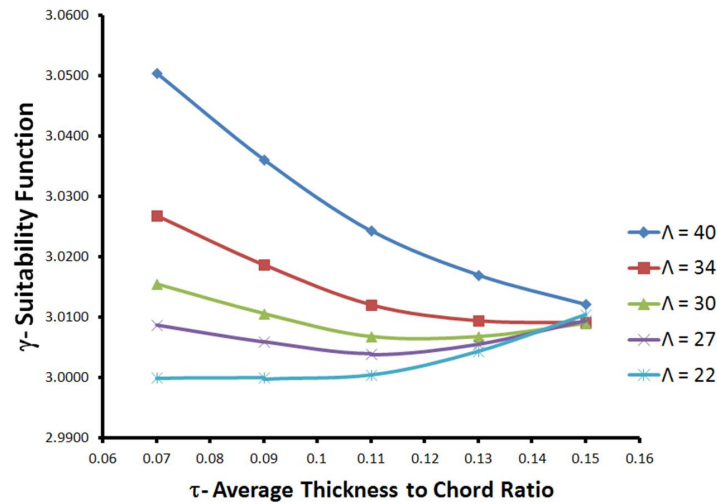


Figure 7.6: γ against τ

7.2 Wing Area Optimization

The wing area affects so many parameters of an aircraft. It determines the wing loading which influences field performance. The wing area determines the lift coefficient which relates to the lift to drag ratio. It therefore influences economic parameters like DOC and fuel per pax per nautical mile. Optimizing wing area is therefore imperative to overall aircraft conceptual design.

7.2.1 S Optimization - Box Wing Aircraft

An optimization procedure was therefore performed to determine the optimum wing area for the box wing using the baseline box aircraft parameters but implementing the opti-

mized value of τ (see Chapter 7.1.1) in all models. Also, from the preceding section fore and aft wing sweep angles of 30 and -22 degrees were chosen as the sweep angles that satisfy the minimization of M_{TOM} , prevention of the effects of drag divergence mach number, mitigating aeroelastic issues for the model and minimizing DOC. Hence, eight wing areas; $240m^2$, $235m^2$, $230m^2$, $224m^2$, $216m^2$, $200m^2$, $190m^2$ and $180m^2$ were arbitrarily selected and their models built the in the optimization tool. Using the macros outlined in Appendix B the optimization tool introduced in Chapter 3 Section 1 was exercised on each model to get results shown in Table 7.1.

Note the high L/D values of the models in Table 7.1. Although the box wing configuration is more efficient than the conventional configuration, in the non-ideal conditions of aircraft operations these high L/D figures could reduce to about 85% to 90% of their value. It emerged that the models with wings areas of $190m^2$ and $180m^2$ could not fly the design range of 4000nm because of an inadequate wing fuel tank capacity as evidenced by their fuel ratios (FR) in column 4 of Table 7.1.

Table 7.1: Box Wing Models Optimization Results

S (m^2)	MTOM (kg)	L/D	FR	DOC/nm	Fuel/pax/nm	TO_{Dist} (m)
240	115979.49	23.55	1.45	21.85	0.0535	1271.04
235	115498.12	23.63	1.39	21.78	0.0531	1291.14
230	114912.59	23.73	1.34	21.70	0.0526	1310.99
224	114135.97	23.84	1.30	21.58	0.0520	1335.15
216	113331.37	23.98	1.21	21.47	0.0514	1372.14
200	111424.70	24.30	1.08	21.19	0.0500	1451.34
190	110172.94	24.50	0.99	21.01	0.0490	1506.90
180	109076.21	24.71	0.90	20.84	0.0482	1570.84

where the fuel ratio, FR, is given by $FR = \frac{BF}{AFC}$

DOC of Table 7.1 is in fiscal year 2007 USD.

A combined graph of L/D, fuel per pax per nautical mile, TO_{Dist} , DOC and M_{TOM} against wing area is shown in Fig 7.7. The models with wing areas of $190m^2$ and $180m^2$ can not fly the design range of 4000nm and are highlighted by the shaded portion of Fig 7.7. Thus, it would seem that the model with $S = 200m^2$ is the optimum wing area because it offers the highest L/D, lowest fuel per pax per nautical mile, lowest M_{TOM} and DOC. This is substantiated by Fig 7.8 which is a plot of a function beta, β , against wing area S where the lower the value of β the more optimum the model.

Beta is given by the equation below.

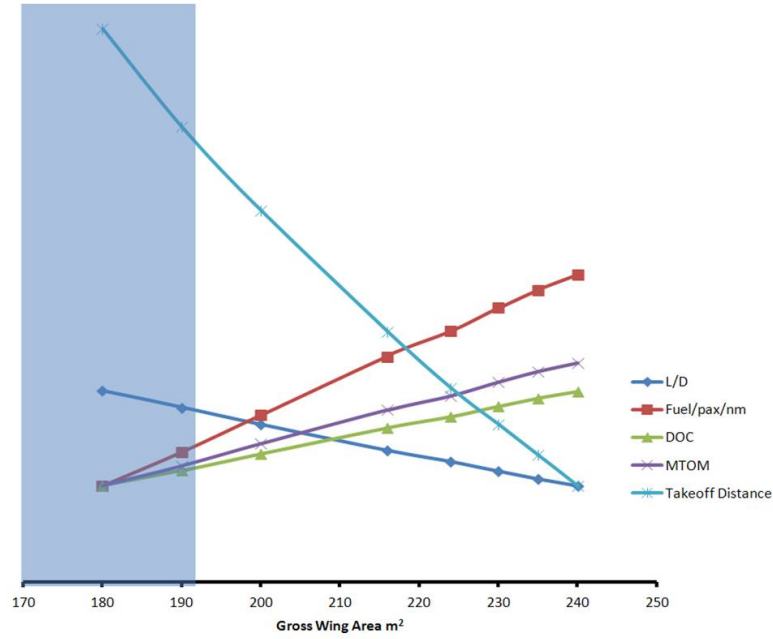


Figure 7.7: Optimization parameters against S - Box Wing Aircraft

$$(\beta)_S = \left(\frac{M_{TOM}}{(M_{TOM})_{min}} \right) + \left(\frac{(L/D)_{min}}{L/D} \right) + \left(\frac{DOC}{(DOC)_{min}} \right) + \left(\frac{Fuel/pax/nm}{(Fuel/pax/nm)_{min}} \right) + \left(\frac{TO_{Dist}}{(TO_{Dist})_{min}} \right) \quad (7.2.1)$$

where

$$M_{TOM} = M_{TOM}(S),$$

$$L/D = L/D(S),$$

$$DOC = DOC(S),$$

$$Fuel/pax/nm = Fuel/pax/nm(S),$$

and

$$TO_{Dist} = TO_{Dist}(S),$$

All factors in Eqn 7.2.1 affect each other directly and indirectly as is usual with aircraft design. The computation was performed simultaneously hence no duplication or layering of the process. The benefit of Eqn 7.2.1 is that the best of all the variables as a function of wing area can be represented by one function and enables the decision making process with regards to the optimum wing area.

However, plotting the payload range diagrams of the models show how their useful loads vary with wing area, Fig 7.9. The useful load is defined here as the difference between the aircraft mass (maximum payload) and aircraft mass (maximum fuel). The higher the useful load the more flexible an aircraft would be. By evaluating the useful loads of the models as a fraction of their maximum payloads and presenting it alongside the same parameter of the similar in-service aircraft (the B767-200 and A310-300), a practical sense of what is optimum becomes clear. These values are presented in the bar chart of Fig

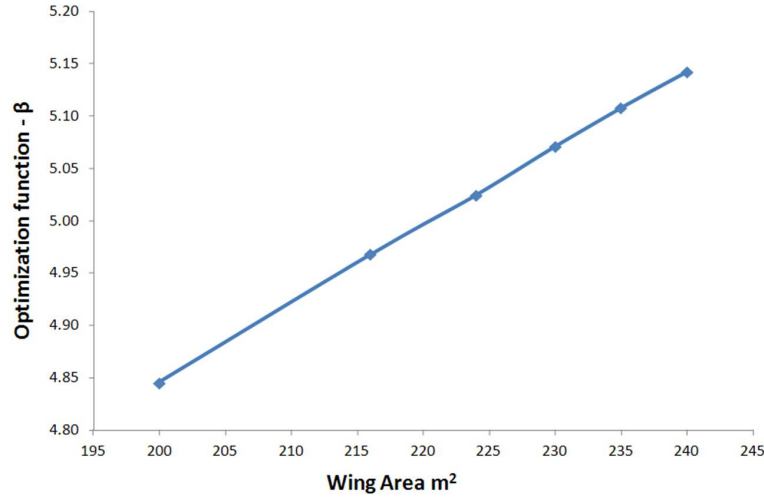


Figure 7.8: Optimization function β against S - Box Wing Aircraft

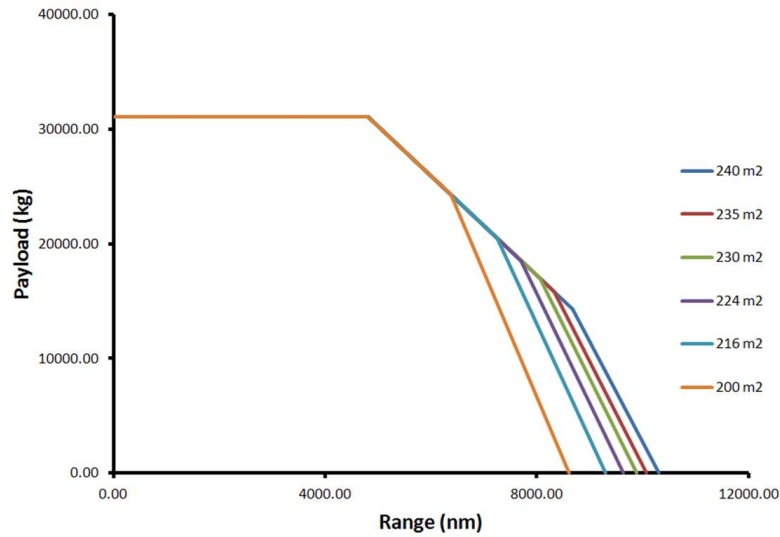


Figure 7.9: Payload Range Plots for Wing Area Models

7.10. The relationship of the the useful loads to wing areas is given by

$$UsefulLoad = 0.008S - 1.3823 \quad (7.2.2)$$

Equation 7.2.2 has a coefficient of determination R^2 of 0.999 and by inputting the average useful loads of the B767-200 and A310-300 the optimum wing area, S , becomes $224m^2$.

7.2.2 S Optimization - Conventional Cantilever Aircraft

A similar optimization procedure to the preceding section was performed to determine the optimum wing area for the conventional cantilever aircraft using the baseline aircraft parameters but implementing the optimized value of τ (see Chapter 8.1.2) in all models.

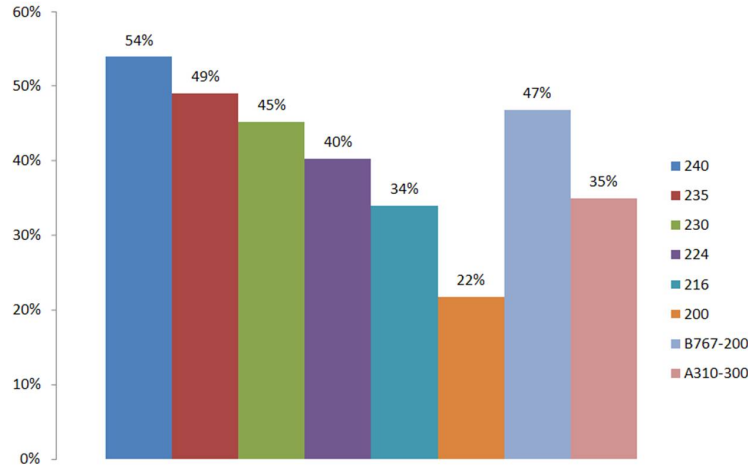


Figure 7.10: Useful Load for Wing Area Models

Also, from section 8.1.2 the wing sweep angle of 30 degrees was chosen as the sweep angles that satisfies the minimization of M_{TOM} and prevention of the effects of drag divergence mach number for the models. Hence, ten wing areas; $240m^2$, $235m^2$, $230m^2$, $224m^2$, $216m^2$, $200m^2$, $190m^2$, $180m^2$, $170m^2$ and $160m^2$ as in the preceding section were selected and their models built the in the optimization tool. Using the macros outlined in Appendix B the optimization tool introduced in Chapter 3 Section 1 was exercised on each model to get results shown in Table 7.2. Note that FR and DOC are as defined in the preceding section.

Table 7.2: Conventional Aircraft Models Optimization Results

S (m^2)	MTOM (kg)	L/D	FR	DOC	Fuel/pax/nm	TO_{Dist} (m)
240	125942.70	17.46	1.93	23.01	0.0750	1485.98
235	125355.08	17.54	1.84	22.92	0.0744	1510.60
230	124830.74	17.63	1.79	22.82	0.0738	1535.43
224	124058.91	17.72	1.73	22.70	0.0730	1565.54
216	123182.65	17.86	1.60	22.55	0.0720	1610.89
200	121263.74	18.13	1.42	22.23	0.0700	1710.22
190	120026.97	18.32	1.31	22.03	0.0687	1780.14
180	118853.07	18.51	1.20	21.83	0.0674	1859.20
170	117610.92	18.70	1.09	21.63	0.0661	1947.38
160	116424.95	18.91	0.98	21.43	0.0649	2048.10

A combined graph of L/D, fuel per pax per nautical mile, takeoff distance, DOC and M_{TOM} against wing area is shown in Fig 7.11. The model with wing area of $160m^2$ could not fly the design range of 4000nm because of an inadequate wing tank fuel capacity and are highlighted by the shaded portion of Fig 7.11. Thus, it seems that the model with $S = 170m^2$ is the optimum wing area because it offers the highest L/D, lowest fuel per pax per nautical mile, lowest M_{TOM} and DOC . This is substantiated by Fig 7.12 which

is a plot of a function beta, β , against wing area S , where as the lower the value of β the more optimum the model.

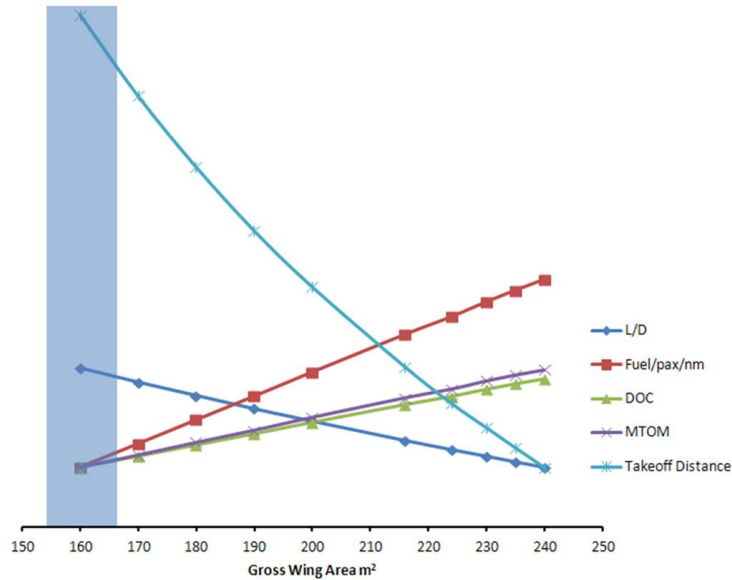


Figure 7.11: Optimization parameters against S - Conventional Configuration

However, plotting the payload range diagrams of models show how their useful loads vary with wing area, Fig 7.13. It shows how the useful load varies from $S = 240m^2$ where due to the large volumetric capacity of the wings there is no capacity left for payload to be carried to the other extreme $S = 160m^2$ where at maximum fuel capacity almost equals maximum payload that can be carried.

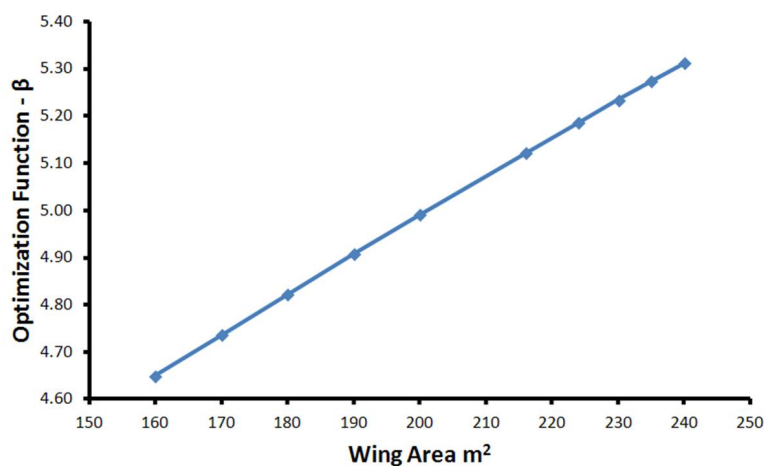


Figure 7.12: Optimization function β against S - Conventional Configuration

The higher the useful load the more flexible is the aircraft. Additionally, and by evaluating the useful loads of the models as a fraction of their maximum payloads and presenting it

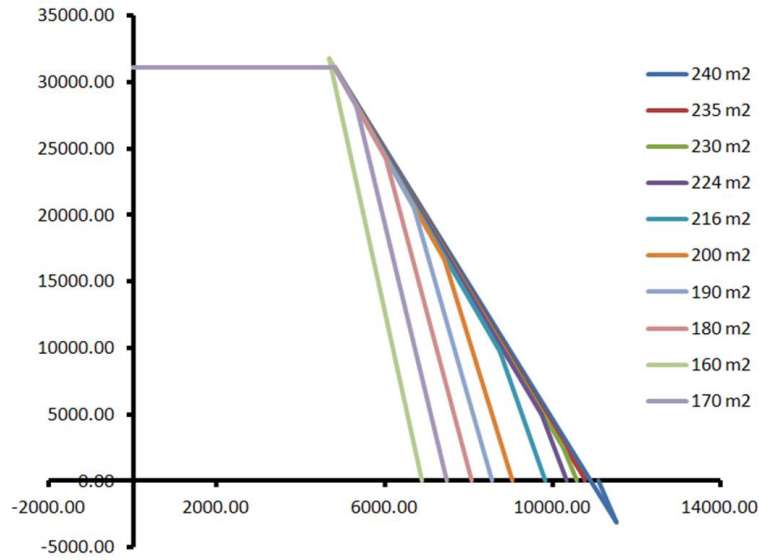


Figure 7.13: Payload Range Plots for Wing Area Models

alongside the same parameter of the equivalent real life aircraft of this study, the B767-200 and A310-300, a practical sense of what is optimum became clear. These values are presented in the bar chart of Fig 7.14. The relationship of the the useful loads to wing areas is given by

$$UsefulLoad = 0.0143S - 2.3589 \quad (7.2.3)$$

Equation 7.2.3 has a coefficient of determination R^2 of 0.9959 and by inputting the average useful loads of the B767-200 and A310-300 the optimum wing area, S , becomes $194m^2$.

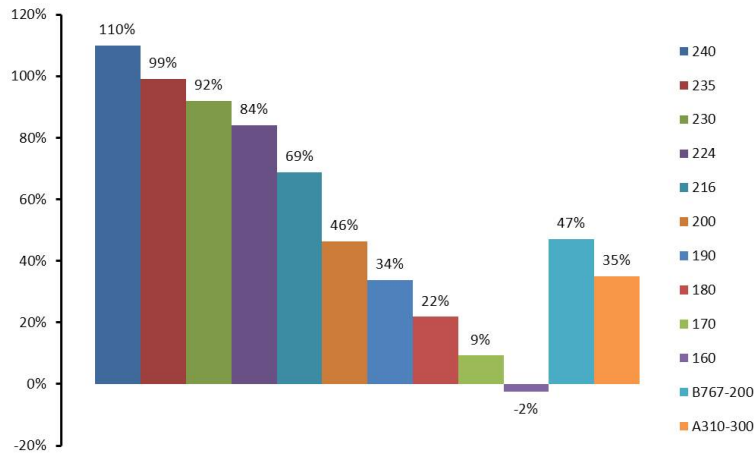


Figure 7.14: Useful Load for Wing Area Models

7.3 Results - Optimum and Baseline Aircraft

The outcomes of the optimization routines and geometrical adjustments to surfaces due to stability and control requirements are presented. The conventional and box wing aircraft are compared to their baseline models.

7.3.1 Conventional Cantilever Aircraft

Table 7.3: Geometric, Weights and Performance Outcomes - Conventional Aircraft

Aircraft Model	Baseline	Optimized	Change (%)
External Dimensions			
AR	9.33	11.39	+22%
S (m^2)	236.64	194.00	-19%
Masses			
$OEMs$ (kg)	69,900.00	55,477.00	-21%
Design payload (kg)	29,700.00	31,050.00	+5%
M_{TOM} (kg)	136,000.00	114,916.00	-16%
Max fuel (kg)	75,500.00	46630.00	-28%
Performance			
LFL (m)	1561	1783	+14%
TFL (m)	1216	1640	+35%
Market Price (2007USD)	142.6m	108.7m	-24%
Fuel/pax/nm (kg)	0.039	0.028	-28%
DOC/nm (USD/nm)	27.48	20.88	-24%
DOC/nm/seat	0.102	0.077	-25%

A comparison of how the optimized conventional cantilever wing aircraft compares with its baseline is presented in Table 7.3. Only parameters that changed are tabled. The process of optimization improved the aspect ratio by 22% even though the wing area reduced by 19%. There was a reduction in the operating empty mass by 21% while design payload increased by 5%. The increase in design payload was because the passenger baggage allowance was increased from 30kg to 40kg in the mass module of the optimization routine for a better reflection of what obtains in practice. The maximum takeoff mass reduced by 16% as did the maximum fuel capacity by 28%. The increase in wing loading caused landing and take-off field lengths to increase by 14% and 35% respectively but this increase still satisfies the constraint of a balanced field length of 2500m. Furthermore, the aircraft's market price reduced by 24% as did the fuel/pax/nm by 28%. The DOC/nm and DOC/nm/seat both improved by 24% and 25% respectively. Thus, the optimized conventional cantilever wing aircraft is a much better aircraft than its baseline. Of significance in the results is the decrease in the M_{TOM} which has a direct relationship with

the aircraft's market price, fuel/pax/nm, DOC/nm and DOC/nm/seat⁷⁷.

Although the results shown in Table 7.3 are optimistic, the improved aspect ratio of 11.39 is rather high for aircraft of this category. What is typical are values below 10. The implications of this over-optimized wing include a rather heavy wing. Furthermore, for a medium range transport aircraft the high aspect ratio of 11.39 also means that there would be aeroelastic problems to contend with. This issue can be mitigated by adjusting the wing area and geometry but for the purpose of this optimization exercise this aspect was left for future investigations.

7.3.2 Box Wing Aircraft

Table 7.4: Geometric Weights and Performance Outcomes - Box Wing Aircraft

Aircraft Type	Baseline	Optimized	Change (%)
External Dimensions			
Fore wing AR	11.95	12.62	+6%
Aft wing AR	11.95	12.62	+6%
Fore wing S (m^2)	118.32	112.00	-6%
Aft wing S (m^2)	118.32	112.00	-6%
Fore wing Λ ($^\circ$)	40	30	-25%
Masses			
$OEMs$ (kg)	68,200.00	57605.00	-16%
Design payload (kg)	29,700.00	31,050.00	+6%
M_{TOM} (kg)	126,670	114240.00	-10%
Max fuel (kg)	44,000.00	37692.00	-15%
Performance			
LFL (m)	1648	1615	-2%
TFL (m)	1204	1336	+11%
Market Price (2007USD)	152.5m	121.9m	-20%
Fuel/pax/nm (kg)	0.036	0.024	-34%
DOC/nm (USD/nm)	24.03	20.34	-15%
DOC/nm/seat	0.089	0.075	-14%

A comparison of how the optimized box wing aircraft compares with its baseline is presented in Table 7.4. Only parameters that changed are presented. The aspect ratios improved by only 6% while there was a similar 6% reduction in gross wing areas. The fore wing sweep angle reduced from 40° to 30° . Operating empty mass reduced by 16% while payload increased by 6%. The increase in payload is as explained in Section 7.3.1. Maximum takeoff weight and maximum fuel capacity reduced by 10% and 15% respectively. There was a marginal 2% reduction in landing field length but takeoff field length increased by 11% likely due to the reduction in wing area. Significant reductions of

20% for aircraft market price and 34% for fuel/pax/nm makes the optimized box wing much better than the baseline. This is substantiated by the 15% and 14% reductions in DOC/nm and DOC/nm/seat respectively.

7.4 Chapter Summary

This chapter has presented the optimizations performed on the box wing and conventional cantilever wing aircraft using design variables of wing sweep angle, average thickness to chord ratio and wing area. The objective functions were minimization of M_{TOM} , DOC, and Fuel/pax/nm. The results of the deterministic and stochastic search methods were subsequently used to perform response surface analysis and regression analysis. The results indicate that the optimized aircraft are better than their baselines.

Stability and Control

8.1 Introduction

Stability and control is concerned with the control actions required to establish an aircraft in equilibrium and with the characteristics required to ensure that the aircraft remains in equilibrium⁴. Unlike box wing aircraft, stability and control issues of conventional cantilever aircraft are generally well explored in literature. It was therefore instructive to investigate how the conventional cantilever aircraft compares with the box wing aircraft. Thus, following the procedure outlined in Fig 8.1 the stability and control of the optimized box wing was analysed and then compared with the conventional cantilever wing aircraft designed in Chapter 7. Accordingly, mass statements from the performance module were used to produce the mass and cg situations of both aircraft. Also, aerodynamic data along with engine performance data were generated using the optimized aircraft as reference. The mass and cg situations were used to produce the aircraft inertia statements. The aircraft models were thereafter built in J2. Trim and response analyses were thereafter performed and the results analyzed. If the models satisfied the requirements given in Moorhouse and Woodcock⁵⁹ it was then inputted in the pilot opinion chart⁴. Otherwise, the model was modified and the process repeated again. Only longitudinal trim, short period oscillation and phugoid modes analyses were investigated.

8.2 Static Margin

An aircraft's longitudinal stability is described in terms of the stability margin which quantifies how much stability the aircraft has over and above zero or neutral stability. The static stability of an aircraft is determined in terms of the distance between the aircraft's neutral point and its *cg* as a percentage of the mean aerodynamic chord. While it is fairly easy to establish an aircraft's *cg*, the difficulty is in determining the aircraft's neutral point. Factors such as thrust line of action with respect to *cg* and indirect power effects caused by the induced flow associated with the intake and exhaust of the engine, affect the neutral point position on an aircraft⁴. Wing sweep, downwash and aircraft geometry (which influences the downwash at the tailplane) also affect the position of an aircraft's neutral point. Thus, a fully representative equation to calculate the neutral

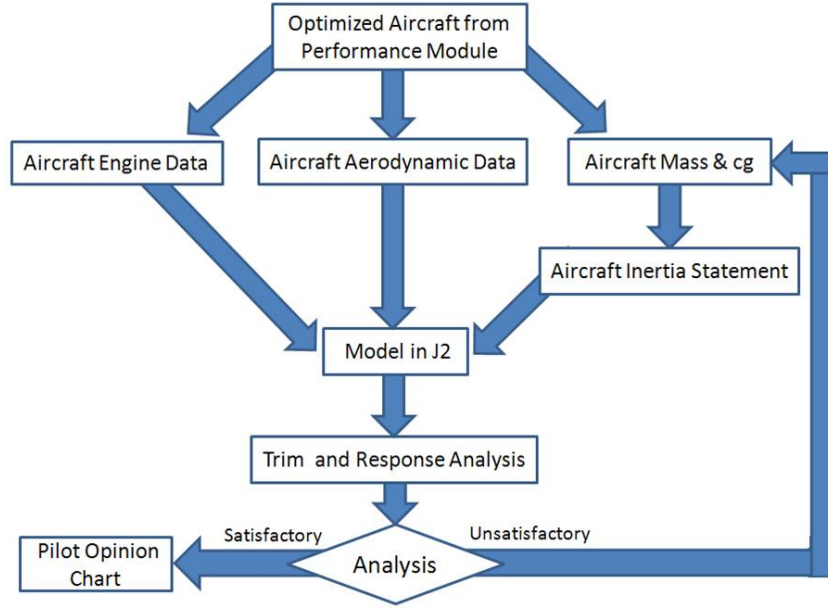


Figure 8.1: Stability and Control Evaluation Schematic

point of an aircraft is difficult to develop but simple equations have been developed for conventional cantilever aircraft.

8.2.1 Neutral Point - Conventional Cantilever Aircraft

Cook⁴ gives a simplified equation for the location of the control fixed neutral point as:

$$h_n = h_0 - \bar{V}_T \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) \quad (8.2.1)$$

Torenbeek's⁷⁰ equation for the control fixed neutral point is more elaborate than Cook's⁴ as the effects of nacelles on the location of the aerodynamic center and fuselage effects can be added via C_{L_α} . Torenbeek's⁷⁰ for the control fixed neutral point is given as:

$$\frac{x_n}{\bar{c}} = \frac{x_{ac}}{\bar{c}} + \frac{C_{L_{h\alpha}}}{C_{L_\alpha}} \left(1 - \frac{d\varepsilon_h}{d\alpha} \right) \frac{S_h l_h}{S \bar{c}} \frac{q_h}{q} \quad (8.2.2)$$

where

x_n = Neutral point position on mean aerodynamic chord

x_{ac} = Aerodynamic center position on mean aerodynamic chord

$C_{L_{h\alpha}}$ = Tailplane lift curve slope

C_{L_α} = Lift curve slope

S_h = Tailplane area

l_h = Tail moment arm

q_h = Dynamic pressure at tailplane

ε_h = Downwash angle at Tailplane

$C_{L\alpha}$ is given as:

$$C_{L\alpha} = \left(C_{L\alpha} \right)_{wf} + C_{L_{h\alpha}} \left(1 - \frac{d\varepsilon_h}{d\alpha} \right) \frac{S_h}{S} \frac{q_h}{q} \quad (8.2.3)$$

where

$\left(C_{L\alpha} \right)_{wf}$ = Wing fuselage combination lift curve slope

Thus, Torenbeek's⁷⁰ equation was used to determine the neutral point of the conventional cantilever wing aircraft implementing a 5%⁷⁰ static margin. This was done using the optimized mass and cg obtained for the conventional aircraft from the average thickness to chord ratio and wing area studies in Chapter 7. Fig 8.2 shows the positions of the *OEM* cg and *NP* locations on the mean aerodynamic chord, $\bar{c}$.

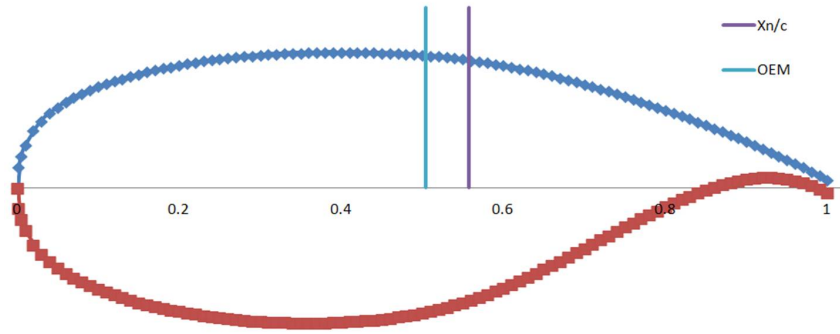


Figure 8.2: Parameter Positions on $\bar{c}$ - Conventional Aircraft

8.2.2 Neutral Point - Box Wing Aircraft

This work's box wing aircraft is based on Prandtl's⁸ best wing system which he describes as a biplane where both wings are of equal span and are joined by tip fins making it a closed system. He gives the condition for minimum induced drag as when there is same lift distribution and same total lift on each of the horizontal wings and butterfly shaped lift distribution on the tip fins. The applicability of Prandtl's best wing system to the box wing configuration is by Munk's⁷ stagger theorem which states that '*The total induced drag of a system of lifting surfaces is not changed when the elements are moved in the streamwise direction provided the lift distribution remains unchanged*'. This can be achieved by optimizing both wings' rigging angles and wing twist to achieve equal forces on both wings. Thus, Prandtl's best wing system suggests neutral stability for the box wing configuration but Frediani's¹² work showed that a box wing can be '*stable in cruise flight, the margin of stability can be controlled and modified and at the same time the lift is equal on the front and rear wings*'.

However, there is not much literature on how to determine the neutral point of box wing aircraft. Roskam's⁵⁸ equation given as:

$$\bar{X}_{ac_A} = \frac{\bar{X}_{ac_{wf}} - \frac{C_{L\alpha_c}}{C_{L\alpha_{wf}}} \eta_c \frac{S_c}{S} \bar{X}_{ac_c} \left(1 + \frac{d\varepsilon_c}{d\alpha}\right) + \frac{C_{L\alpha_h}}{C_{L\alpha_{wf}}} \eta_h \frac{S_h}{S} \bar{X}_{ac_h} \left(1 - \frac{d\varepsilon}{d\alpha}\right)}{1 + \frac{C_{L\alpha_c}}{C_{L\alpha_{wf}}} \eta_c \frac{S_c}{S} \left(1 + \frac{d\varepsilon_c}{d\alpha}\right) + \frac{C_{L\alpha_h}}{C_{L\alpha_{wf}}} \eta_h \frac{S_h}{S} \left(1 - \frac{d\varepsilon}{d\alpha}\right)} \quad (8.2.4)$$

is for a three surface configuration aircraft. Similarly, Stinton's⁷⁸ and Phillips⁷⁹ equations for neutral point computation are for conventional and canard configurations respectively. NASA's⁸⁰ Report CR - 132462 gives the box wing neutral point position in terms of the mean aerodynamic chord as

$$h_n = h_{o_1} + \frac{\left(l_2 + h_{o_2} \bar{c}_2 - h_{o_1} \bar{c}_1\right) \left(1 - \frac{d\varepsilon}{d\alpha}\right) \frac{C_{L\alpha_2}}{C_{L\alpha_1}}}{1 + \left(1 - \frac{d\varepsilon}{d\alpha}\right) \frac{C_{L\alpha_2}}{C_{L\alpha_1}}} \quad (8.2.5)$$

where

h_{o_1} = mean aerodynamic position on fore wing

h_{o_2} = mean aerodynamic position on aft wing

l_2 = distance between the leading edges of the fore and aft wings

$\bar{c}_1$ = fore wing mean aerodynamic chord

$\bar{c}_2$ = aft wing mean aerodynamic chord

$C_{L\alpha_1}$ = fore wing lift curve slope

$C_{L\alpha_2}$ = aft wing lift curve slope

However, this equation is dimensionally inconsistent and so could not be used to determine the neutral point position of the box wing in this investigation. Hence, an equation had to be derived to this purpose.

8.2.3 Box Wing Aircraft Neutral Point Derivation

A fully representative general pitching moment equation is difficult to develop since it is dependent on the geometry of the aircraft and so many other factors. However, it is possible to develop a simple approximation to the pitching moment equation, which is sufficiently representative for preliminary studies and which provides considerable insight into the basic requirements for box wing aircraft static stability and trim.

The general requirement for longitudinal stability may be expressed as :

$$\frac{dC_m}{d\alpha} < 0 \quad (8.2.6)$$

To develop a simple pitching moment equation, a model was defined showing only the normal forces and pitching moments acting on the aircraft. The aircraft was assumed to

be in un-accelerated level flight where thrust and drag are in equilibrium and act at the center of gravity. It was also assumed that changes in this equilibrium were insignificant for small disturbances in incidence. Therefore, from the foregoing, small disturbances in incidence caused significant changes in lift forces and pitching moments only. The model used to derive the pitching moment equation is shown in Fig 8.3.

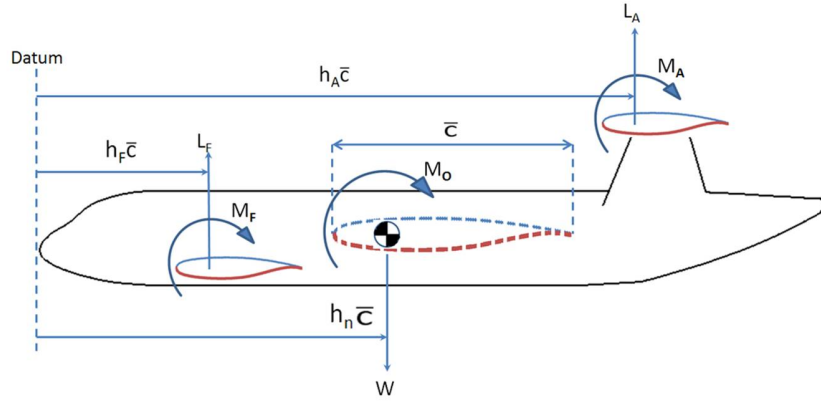


Figure 8.3: Simple Pitching Moment Model

Referring to Fig 8.3, the lift forces L_F and L_A , pitching moments M_o , M_F and M_A are assumed to act at the aerodynamic centres. An expression for the total pitching moment M about the cg may therefore be written as:

$$M = M_o + M_F + M_A + L_F X_1 - L_A X_2 \quad (8.2.7)$$

where $X_1 = h_n \bar{c} - h_F \bar{c}$ and $X_2 = h_A \bar{c} - h_n \bar{c}$. Note that $\bar{c} = c_F + c_A$ where c_F is the fore wing mean aerodynamic chord and c_A is the aft wing mean aerodynamic chord.

In coefficient form the above equation may be written as:

$$qSC_m \bar{c} = qSC_{m_o} \bar{c} + qS_F C_{m_F} c_F + qS_A C_{m_A} c_A + qS_F C_{l_F} X_1 - qS_A C_{l_A} X_2 \quad (8.2.8)$$

Dividing through by $qS\bar{c}$

$$C_m = C_{m_o} + \frac{S_F C_{m_F} c_F}{S \bar{c}} + \frac{S_A C_{m_A} c_A}{S \bar{c}} + \frac{S_F C_{l_F} X_1}{S \bar{c}} - \frac{S_A C_{l_A} X_2}{S \bar{c}} \quad (8.2.9)$$

Considering that $S_F = S_A = 0.5S$ and that $\frac{S_F X_1}{S \bar{c}} = 0.5 \frac{X_1}{\bar{c}} = V_F$, $\frac{S_A X_2}{S \bar{c}} = 0.5 \frac{X_2}{\bar{c}} = V_A$ Equation 8.2.9 becomes

$$C_m = C_{m_o} + 0.5 C_{m_F} \frac{c_F}{\bar{c}} + 0.5 C_{m_A} \frac{c_A}{\bar{c}} + V_F C_{l_F} - V_A C_{l_A} \quad (8.2.10)$$

The aft wing lift coefficient C_{l_A} , may be expressed as:

$$C_{l_A} = a_o + a_1 \alpha_A + a_2 \eta + a_3 \beta_\eta \quad (8.2.11)$$

where a_o, a_1, a_2 and a_3 are constant aerodynamic coefficients.

Neglecting aft wing elevator angle and elevator trim tab angle, Equation 8.2.11 becomes

$$C_{lA} = a_o + a_1 \alpha_A \quad (8.2.12)$$

The angle of attack of the aft wing is given by:

$$\alpha_A = \alpha - \varepsilon + \eta_A \quad (8.2.13)$$

For small disturbances ε is a function of wing-body incidence α only

$$\alpha - \varepsilon = \alpha \left(1 - \frac{d\varepsilon}{d\alpha} \right) = \frac{C_{lF}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) \quad (8.2.14)$$

Equation 8.2.13 becomes

$$\alpha_A = \frac{C_{lF}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + \eta_A \quad (8.2.15)$$

Substituting Equation 8.2.15 in Equation 8.2.12 gives

$$C_{lA} = a_o + a_1 \left(\frac{C_{lF}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + \eta_A \right) \quad (8.2.16)$$

Similarly, by substitution Equation 8.2.10 becomes

$$C_m = C_{mo} + 0.5 C_{mF} \frac{c_F}{\bar{c}} + 0.5 C_{mA} \frac{c_A}{\bar{c}} + V_F C_{lF} - V_A \left(a_o + a_1 \left(\frac{C_{lF}}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + \eta_A \right) \right) \quad (8.2.17)$$

Rearranging Equation 8.2.17 gives

$$C_m = C_{mo} + 0.5 C_{mF} \frac{c_F}{\bar{c}} + 0.5 C_{mA} \frac{c_A}{\bar{c}} + V_F C_{lF} - V_A \left(a_o + \frac{C_{lF} a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) + a_1 \eta_A \right) \quad (8.2.18)$$

For box wing aircraft $L_F = L_A$ therefore $C_{lF} = C_{lA} = C_l$. Thus, differentiating the above equation with respect to C_l and noting that C_{mo} , C_{mF} , C_{mA} , c_F and c_A are by definition constants, Equation 8.2.18 becomes:

$$\frac{dC_m}{dC_l} = V_F - V_A \frac{a_1}{a} \left(1 - \frac{d\varepsilon}{d\alpha} \right) - V_A a_1 \frac{d\eta_A}{dC_l} \quad (8.2.19)$$

For controls fixed stability:

$$\frac{d\eta_A}{dC_l} = 0 \quad (8.2.20)$$

Thus, Equation 8.2.19 becomes:

$$\frac{dC_m}{dC_l} = V_F - V_A \frac{a_l}{a} \left(1 - \frac{d\varepsilon}{d\alpha}\right) \quad (8.2.21)$$

By definition, the neutral point of an aircraft is the center of gravity position that gives

$$\frac{dC_m}{dC_l} = 0 \quad (8.2.22)$$

And given that

$$V_F = 0.5 \frac{X_l}{\bar{c}} = 0.5 \frac{(h_n \bar{c} - h_F \bar{c})}{\bar{c}} = 0.5(h_n - h_F)$$

and

$$V_A = 0.5 \frac{X_2}{\bar{c}} = 0.5 \frac{(h_A \bar{c} - h_n \bar{c})}{\bar{c}} = 0.5(h_A - h_n)$$

Equation 8.2.21 can be rearranged to give

$$h_n = \frac{h_F + h_A \frac{a_l}{a} \left(1 - \frac{d\varepsilon}{d\alpha}\right)}{1 + \frac{a_l}{a} \left(1 - \frac{d\varepsilon}{d\alpha}\right)} \quad (8.2.23)$$

Note the absence of upwash in Equation 8.2.23; it is implicitly accounted for by a which is the wing body lift curve slope. This simplified neutral point equation (Eqn 8.2.23) is sufficient for initial longitudinal static stability investigations for the box wing aircraft. Thus, the neutral point position was computed and it's location, along with the *OEM* and *cg*, on the mean aerodynamic chord, $\bar{c}$, are shown in Fig 8.4.

8.3 Aircraft Mass Statements

Using the optimizations performed in Chapter 7, the mass statements of both aircraft were produced. The fuselage, nacelle, propulsion, landing gear, surface controls and fixed equipment masses of the box wing aircraft were computed using the method given in Jenkinson⁵⁴. The fore and aft wing, tip fin and tail fin masses were computed using the algorithm derived in Chapter 4. Mission fuel was obtained from the Breguet range equation given by Matthews⁸¹ as:

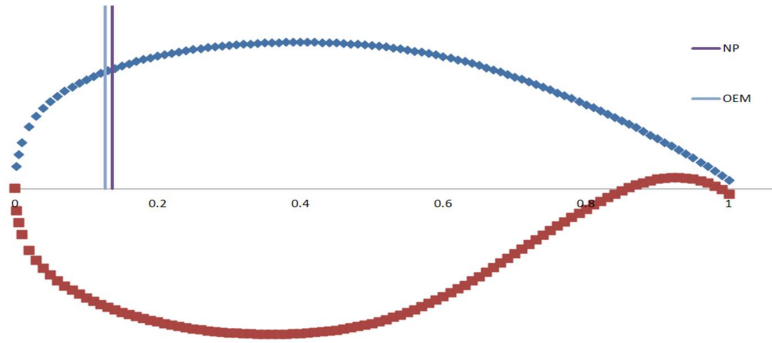


Figure 8.4: Parameter Positions on $\bar{c}$ - Box Wing Aircraft

$$R = \frac{L}{D} \left(\frac{CruiseSpeed}{sfc} \right) \ln \left(\frac{W_o}{W_1} \right) \quad (8.3.1)$$

where

W_o = Initial aircraft mass

W_1 = Final aircraft mass

Wing fuel volume to accommodate the mission fuel was cross-checked in the optimization algorithm which had the method by Jenkinson⁸² embedded in it. Using the payload, the *OEM* and M_{TOM} mass statements were produced. Details are in Appendix D.

Similarly, by applying the geometric parameters of the conventional cantilever aircraft, its fuselage, nacelle, propulsion, landing gear, wing, surface controls, tail fin, tail plane and fixed equipment masses were computed using the method given in Jenkinson⁵⁴. The mission fuel was also computed using the Breguet range equation as given in Matthews⁸¹ and the wing fuel volume to accommodate the mission fuel cross-checked in the optimization algorithm using Jenkinson's⁸² method embedded in it. Finally, the *OEM* and M_{TOM} mass statements were produced. Details are in Appendix D.

8.4 Aircraft Inertia Statements

The outcomes of the preceding section were used to compute the aircraft inertia statements. As outlined in Bruhn⁸³ the inertia of each component was first of all determined about its own centroidal axis then about the axes of the aircraft. Thus the inertia mass statements for both aircraft at *OEM*, *OEM* plus 33% payload, *OEM* plus 66% payload and M_{TOM} were produced. The inertia statements for *OEM* plus 33% payload, *OEM* plus 66% payload were prepared for the trim and response analysis to be performed later. Details are in Appendix E.

For conventional aircraft:

$$I_{xx} + I_{yy} \approx I_{zz} \quad (8.4.1)$$

Which is as stated in Bruhn⁸³. For the box wing aircraft:

$$I_{xx} + I_{yy} > I_{zz} \quad (8.4.2)$$

8.5 Aircraft Aerodynamic and Engine Data

As a prelude to the stability and control analysis using J2 aircraft dynamic software, it was necessary to produce aerodynamic data with which the software would perform the analysis. The following were required:

1. Fore wing lift coefficient variation with α and elevon deflection.
2. Aft wing lift curve slope variation with α and elevator deflection.
3. Fore wing trim drag variation as a function of angle of attack and elevon deflection.
4. Aft wing trim drag variation as a function of angle of attack and elevator deflection.
5. Aircraft pitching moment as a function of aft wing angle of attack and elevator and elevon deflection.
6. Engine thrust as a function of Mach number, altitude and engine throttle setting.

Serials 1 to 5 above were initially computed using methods given by Roskam⁸⁴, ESDU74011⁸⁵ and ESDU89029⁸⁶ however due to the complexity and volume of computations required Javafoil⁸⁷ was used after the initial set of computations.

Javafoil⁸⁷ is a software used to analyze airfoils and aircraft models by potential flow and boundary layer analysis and the results from Javafoil⁸⁷ were in agreement with the hand calculations. The engine thrust as a function of mach number, altitude and engine throttle setting was computed using methods given by Yechout et al⁸⁸.

The values of all the above computations for the box wing and conventional cantilever wing aircraft are in Appendix F. The values were used to build the aircraft models in J2 aircraft dynamics software as a prelude to the longitudinal stability and control analysis.

8.6 Trimming and Stability

The object of trimming is to bring the forces and moments acting on the aircraft into a state of equilibrium. That is the condition when the axial, normal and side forces, and the roll, pitch and yaw moments are all zero⁴.

Thus, trimming analysis was performed for the box wing and conventional aircraft models with 33% and 66 % payload. The analysis was performed for several points within a speed

range of 0 to 240 m/s and altitude range of 0 to 31,000ft. This analysis was intended to show the points in this envelope at which the aircraft could be trimmed longitudinally. Trimming devices used were the elevators for the conventional aircraft and the elevators and elevon (elevator on the forward wing) for the box wing. For the box wing the elevator and elevon work in opposition. Furthermore, the elevon's convention is opposite that of the elevator meaning up is positive and down is negative. The results of these analyses for both aircraft with 33% payload is shown and discussed in subsequent paragraphs while the results for both aircraft at 66% payload is in Appendix G. The sign conventions used are shown in Fig 8.6.

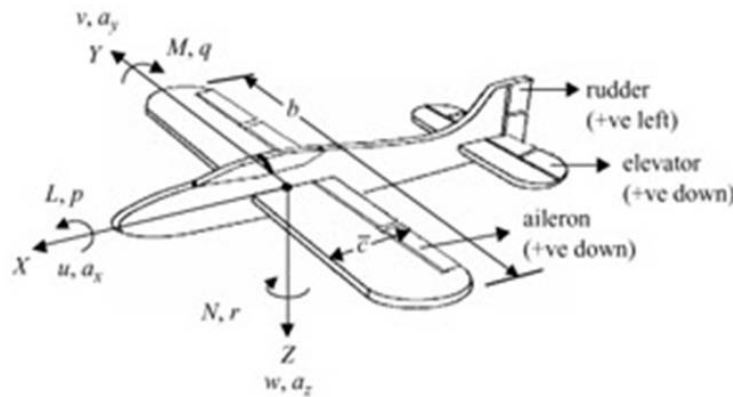


Figure 8.5: Axes and Sign Conventions³

8.6.1 Trim Analysis - Conventional Aircraft

Fig 8.7 is a graph of the trimming analysis for the conventional cantilever wing aircraft. On the y-axis on the left is the aircraft's angle of attack in degrees while on the y-axis on the right is the elevator deflection angle also in degrees. The x-axis displays the true air speed of the vehicle in kts. The speed range displayed is that for which the aircraft is flyable, ie above stall speed at any altitude. The angle of attack is indicated by the red dots while the elevator deflection by the blue dots. Multiple dots on the same speed mark represents different altitudes. Fig 8.7 shows that as speed increases the angle of attack of the aircraft reduces from a maximum of about 16 degrees at 200 knots to -0.5 degrees at about 460 knots. The elevator deflection increases from a minimum of -3.4 degrees at 150 knots to about 0.8 degrees at 460 knots. Thus, the trend of the angle of attack and the elevator is opposite each other and this is what occurs in practice.

The points in the envelope at which the model can be theoretically trimmed is shown graphically in Fig 8.8. On the y-axis is altitude in feet and on the x-axis is true air speed in kts. Thus this model cannot be trimmed at speed below 170 knots in altitudes from 0

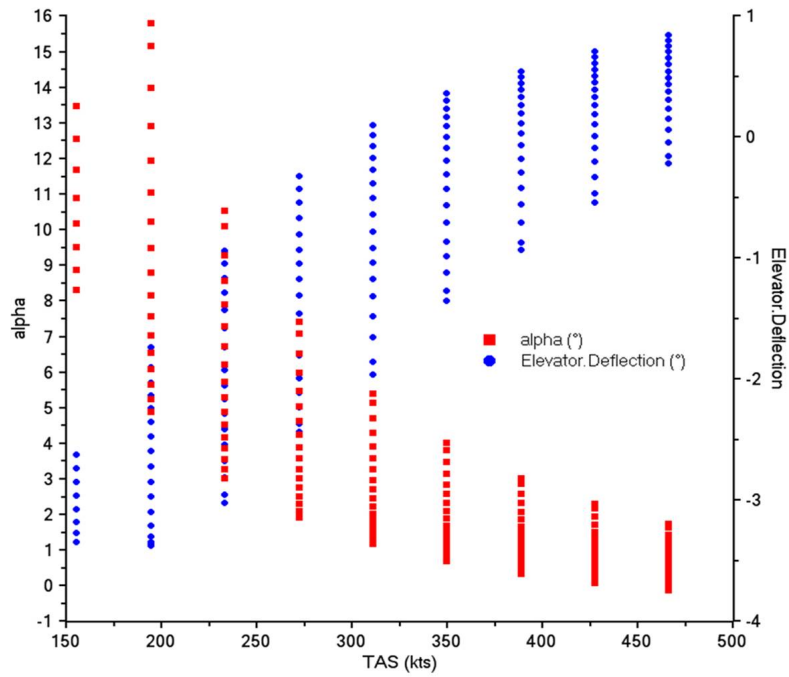


Figure 8.6: AoA and Elevator Deflection to Trim - 33% payload

to 31,000ft.

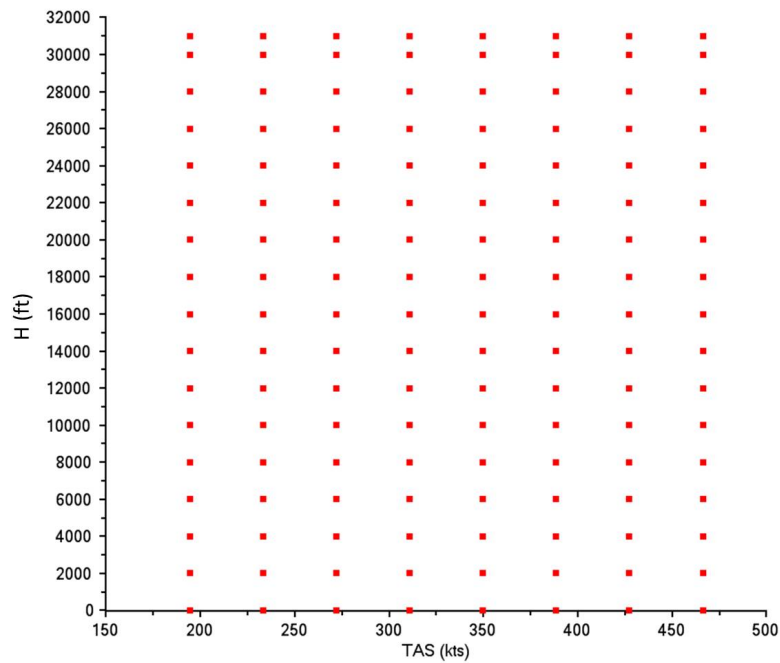


Figure 8.7: Flight Envelope Achievable - 33% payload

8.6.2 Trim Analysis - Box Wing Aircraft

Fig 8.9 shows the trim analysis conducted for the box wing aircraft. The left y-axis shows the angle of attack in degrees while the left y-axis shows the elevon deflection in degrees. The x-axis shows the true air speed in kts. The red dots indicate the angle of attack and the blue dots the elevon deflection. The trend in this graph is not as obvious as in the preceding graph. However, the red dots show a reduction in angle of attack with increase in air speed from about 18 degrees at 230 knots to -1.5 at 460 knots. The elevon deflection, indicated by the blue dots, shows its movement from about -7 degrees at 230 knots to 1.5 degrees at speed.

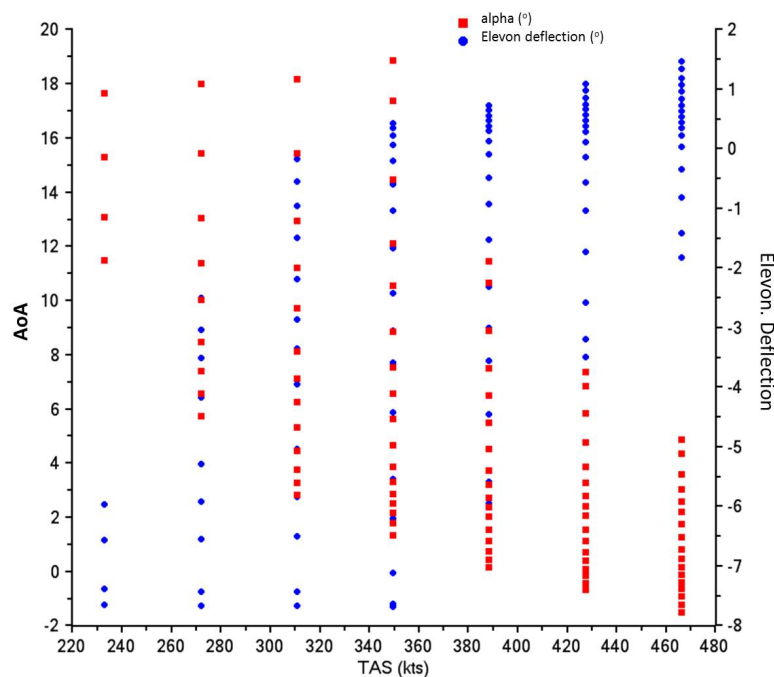


Figure 8.8: Box Wing AoA and Elevon Deflection to Trim - 33% payload

Fig 8.10 shows the same trim analysis conducted for the box wing aircraft but here the left y-axis shows the angle of attack in degrees while the left y-axis shows the elevator deflection in degrees. The x-axis shows the true air speed in kts. The red dots indicate the angle of attack and the blue dots the elevator deflection. Here, the elevator deflection, indicated by the blue dots, shows its movement from about 4.5 degrees at 230 knots to -1.6 degrees at speed. The red dots indicate the angle of attack and show a reduction with increase in air speed from about 18 degrees at 230 knots to -1.5 at 460 knots.

Fig 8.11 provides indication of the trends of the elevon and elevator with increase in air speed. The left y-axis shows the elevon deflection in degrees while the y-axis shows the elevator deflection in degrees. The x-axis shows the true air speed in kts. The red dots indicate the elevon and the blue dots the elevator deflection. Clearly and as already elucidated, the elevon and elevator move in opposite directions and the trends are opposite.

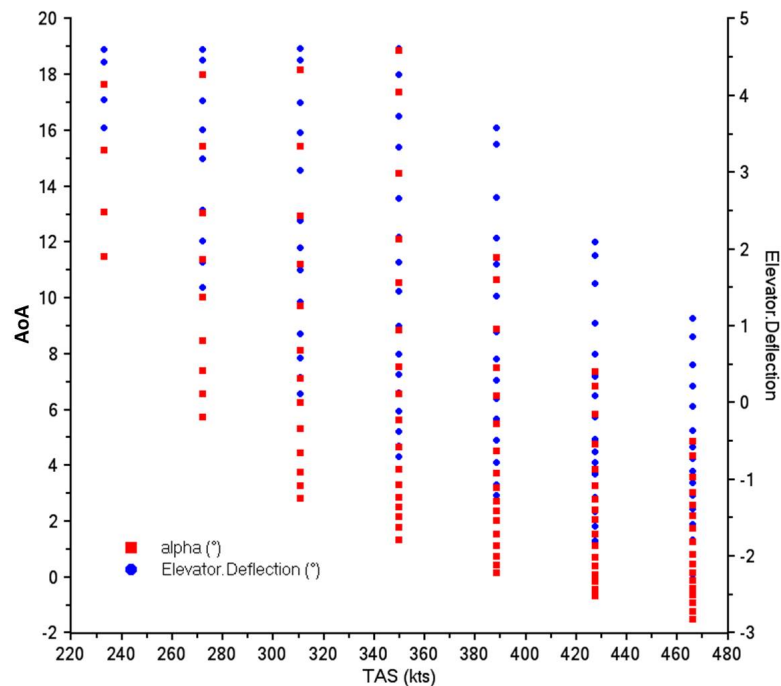


Figure 8.9: Box Wing AoA and Elevator Deflection to Trim - 33% payload

As airspeed increases, the elevon moves from negative to positive within a range of -7.8 degrees to 1.4 degrees, while the elevator moves from positive to negative.

Fig 8.12 shows the points in the envelope at which the model can theoretically be trimmed. On the y-axis is altitude in feet and on the x-axis is true air speed in kts. Thus, this model cannot be trimmed at speed below 230 kts and at 230 kts it can be trimmed only at altitudes below 6000ft. At 270 kts it can be trimmed at altitudes below 16,000ft. At 310 knots the model can be trimmed only below 24,000ft. From 350 knots upwards the model can be theoretically trimmed from zero altitude to 31,000ft.

As a basis for comparison, both aircraft were compared while cruising at 31,000ft at Mach 0.8. From Table 8.1 both aircraft were cruising at about the same angle of attack but while the conventional aircraft's wing had a positive angle of attack the box wing's fore wing had a negative angle of attack. At the tailplane and aft wing both had positive angle of attacks.

The fact that for the box wing aircraft the fore wing is at a '*low*' angle and the aft wing a '*high*' angle is line with Bell's¹⁵ highlight that the rear wing induces an upwash on the forward wing, which in turn induces a downwash on the rear wing. Thus, the fore wing's negative angle of attack is to compensate for the increased angle of attack caused by the upwash on it induced by the aft wing. Similarly, the aft wing's rather high angle of attack is to compensate for the reduced angle of attack induced on it by the downwash from the fore wing.

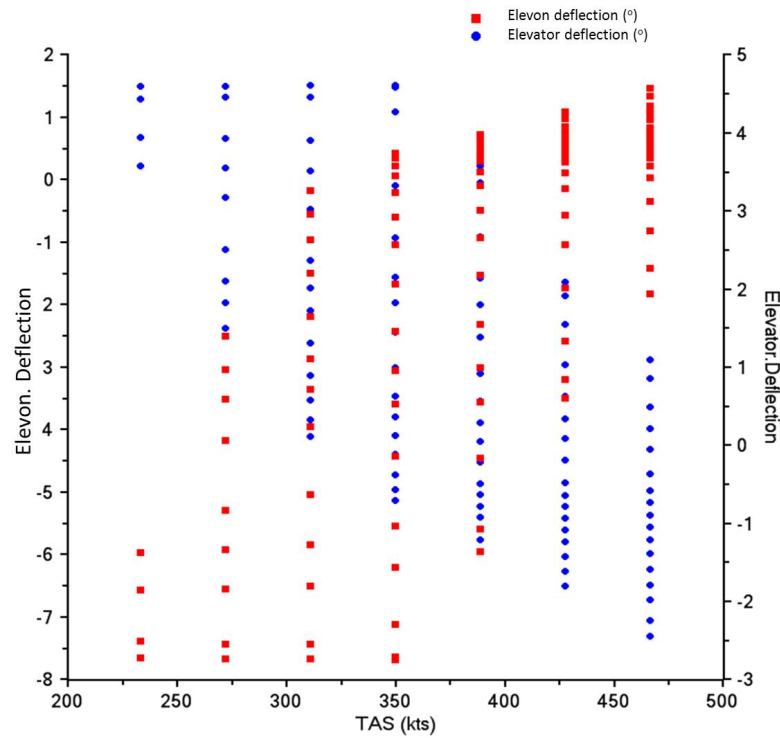


Figure 8.10: Box Wing Elevon and Elevator Deflection to Trim - 33% payload

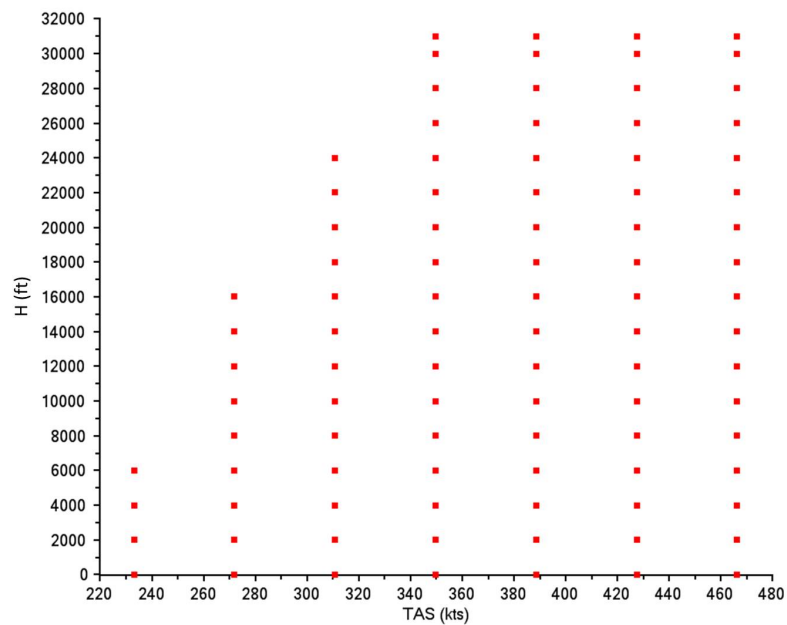


Figure 8.11: Box Wing Flight Envelope Achievable - 33% payload

The trim drag of the conventional aircraft with an elevator angle of -0.22° would be much lower than that of the box wing with elevon and elevator angles of 3.10° and -5.13°

Table 8.1: Aircraft Parameters at Mach 0.8 31,000ft

Type	<i>AoA</i>	Wing/Fore wing <i>AoA</i>	Elevon	Tailplane/Aft wing <i>AoA</i>	Elevator
	($^{\circ}$)	($^{\circ}$)	($^{\circ}$)	($^{\circ}$)	($^{\circ}$)
Conventional	1.70	2.94		1.12	-0.22
Box	1.68	-1.32	3.1	2.10	-5.13

respectively. This suggests that further optimization is required for the box wing as the trim drag suggested by this simulation could reduce the advantage this configuration has over the conventional aircraft.

8.7 Longitudinal Dynamics

The longitudinal dynamics of an aircraft may be likened to a pair of loosely coupled mass-spring-damper systems. The interpretation of the motion of the aeroplane following a disturbance from equilibrium may be made by direct comparison with the behaviour of the mechanical mass-spring-damper. However, the damping and frequency characteristics of the aircraft are not mechanical in origin but dependent on the aerodynamic characteristics⁴.

8.7.1 Short Period Oscillation

Short period oscillation (SPO) mode is typically a damped oscillation in pitch about the lateral axis. The principal variables are angle of attack, pitch rate and pitch attitude. It may be excited by applying a short duration disturbance in pitch to the trimmed aircraft. This is achieved with an elevator pulse sufficiently short so as not to excite the phugoid mode. The short period oscillation evaluations performed were inputted on a longitudinal short period pilot opinion contours chart, Fig 8.13, otherwise called the ‘*thumb print*’ criterion. The ‘*thumb print*’ criterion provides guidance to aircraft designers and evaluators concerning the best combinations of longitudinal short period mode damping and frequency to give good handling qualities. The chart is empirical and is based entirely on pilot opinion but adequate for conceptual design studies. The plot of undamped natural frequency against damping ratio in Fig 8.13 is marked by areas designated as satisfactory, acceptable, poor and unacceptable. By obtaining the undamped natural frequencies and damping ratios of the models, a fair idea of their short period oscillation can be obtained for conceptual design level studies.

8.7.1.1 Short Period Oscillation - Conventional Cantilever Wing Aircraft

The model of the conventional cantilever wing aircraft with 33% payload was simulated at an altitude of 31,000ft at Mach 0.8. After a period of 2 seconds a step input lasting

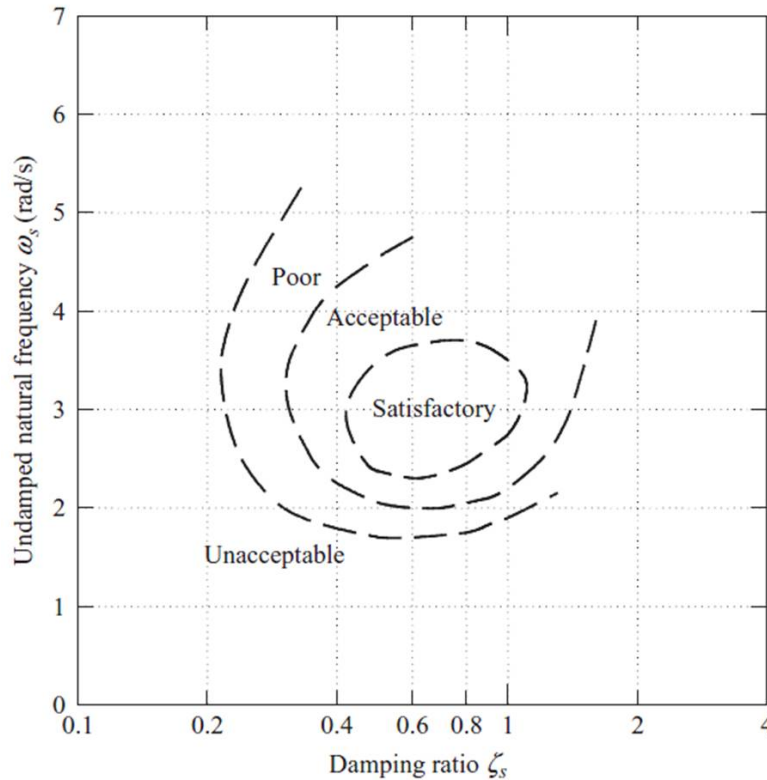


Figure 8.12: Longitudinal Short Period Pilot Opinion Contours⁴

0.02 seconds was inputted by the model's elevator. Fig 8.14 shows the behaviour of the aircraft. The y-axis shows the aircraft's angle of attack in degrees and the x-axis shows the time in seconds. When perturbed, there is an abrupt decrease in angle of attack followed by an overshoot above the trimmed angle of attack then the oscillation dampens out. The overall change in angle of attack during the oscillation is less than a degree and it settles about 14 seconds after the initial perturbation. The computed damping ratio is 0.76 and the undamped natural frequency is 3.12 *rad/s*.

8.7.1.2 Short Period Oscillation - Box Wing Aircraft

The model of the box wing aircraft with 33% payload was also simulated at an altitude of 31,000ft at Mach 0.8. After a period of 2 seconds, a step input lasting 0.02 seconds was inputted moving the model's elevon and elevator in opposite directions. The behaviour of the aircraft is shown in Fig 8.15 where the y-axis is the aircraft's angle of attack in degrees and the x-axis the time in seconds. At the perturbation, there is a much deeper drop in angle of attack compared to the conventional aircraft and the reversal is shallower as the oscillation dampens out. The overall change in angle of attack during the oscillation is about 2.2 degrees and it settled 19 seconds after the initial disturbance. This gives a damping ratio of 0.68 and an undamped natural frequency of 1.8 *rad/s*.

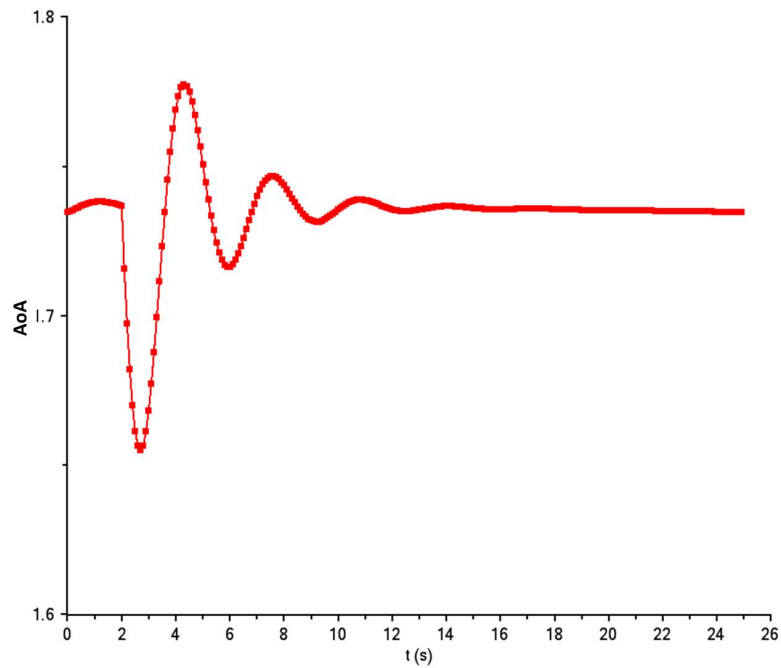


Figure 8.13: Short Period Oscillation - Conventional Aircraft with 33% payload

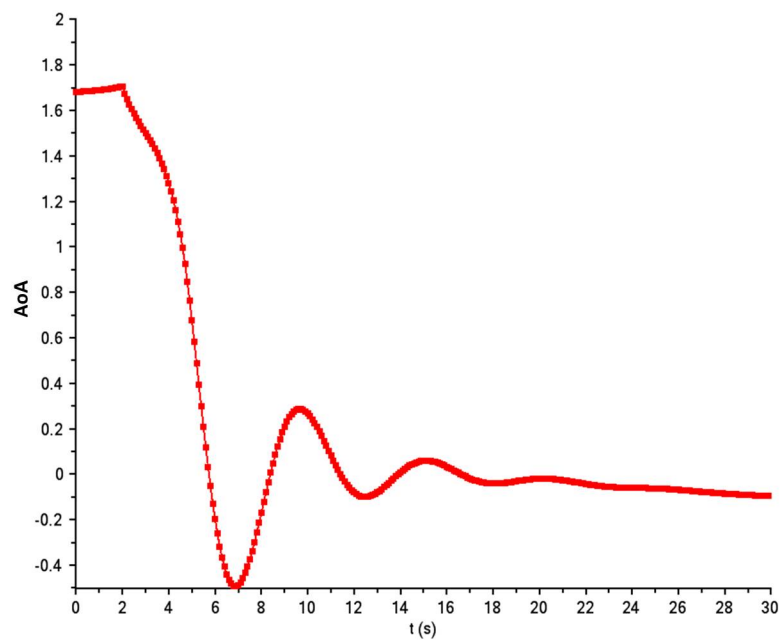


Figure 8.14: Short Period Oscillation - Box Wing Aircraft with 33% payload

8.7.2 Phugoid

The phugoid is a damped harmonic motion resulting in an aircraft flying a gentle sinusoidal flight path about the nominal trimmed height datum. As large inertia and momentum effects are involved the motion is slow. The phugoid mode may be excited by applying

a small speed disturbance to the aircraft in trimmed flight. This is best achieved by applying a small step input to the elevator which will cause the aircraft to fly up, or down, according to the sign of the input⁴.

8.7.2.1 Phugoid - Conventional Cantilever Wing Aircraft

The model of the conventional cantilever wing aircraft with 33% payload was simulated at 31,000ft at Mach 0.8. The behaviour of the aircraft is shown in Fig 8.16 where the left y-axis shows altitude in feet and right y-axis shows the true air speed while the x-axis shows the time in seconds. The trend shown is consistent with a phugoid; the air speed and altitude oscillating in opposition. At the end of the 250 second period the aircraft was on the climb at 724 ft above the start altitude and the speed decreased by 58 kts. The computed damping ratio of 0.001 and undamped natural frequency of 0.07 rad/s are all low and typical of phugoids⁴.

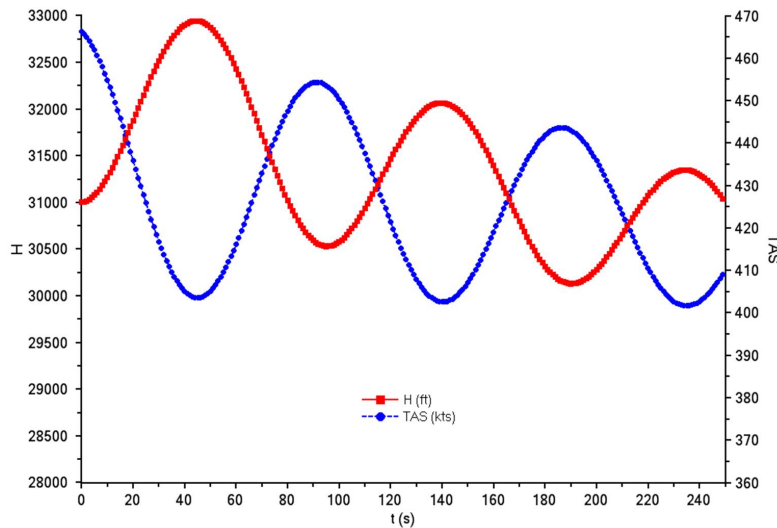


Figure 8.15: Conventional Aircraft Phugoid - 33% payload

8.7.2.2 Phugoid - Box Wing Aircraft

The model of the box wing aircraft with 33% payload was simulated at 31,000ft at Mach 0.8. The behaviour of the aircraft is shown in Fig 8.17 where the left y-axis shows altitude in feet and right y-axis shows the true air speed while the x-axis shows the time in seconds. The trend shown is consistent with a phugoid; the air speed and altitude oscillating in opposition. During the 250 second period the altitude loss was 323 ft and the speed loss 56 kts. These losses are indicative of the gradual damping of the motion. The computed damping ratio of 0.006 and undamped natural frequency of 0.07 rad/s are all low and typical of phugoids⁴.

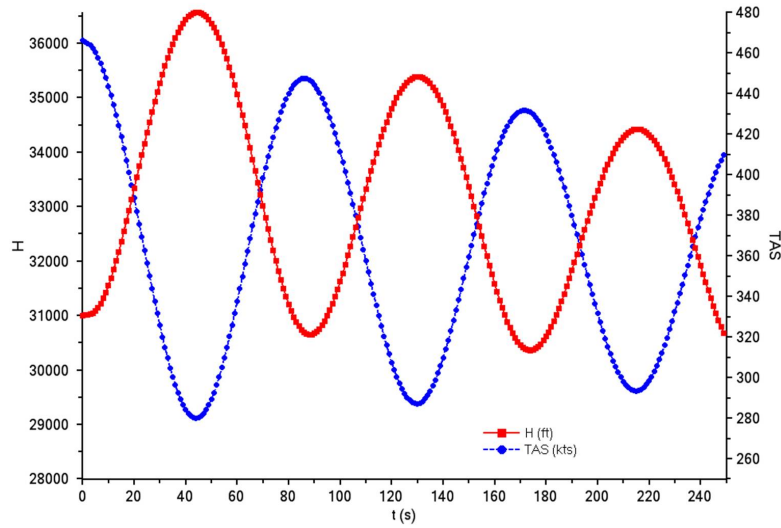


Figure 8.16: Box Wing Phugoid - 33% payload

8.8 Chapter Summary

This chapter has presented the methodology used to investigate the longitudinal stability and control aspects of the conventional cantilever wing and box wing aircraft. The mass statements of the optimum aircraft were used to determine their static stability. For the box wing an equation was derived for its static margin as conventional aircraft equations do not apply to box wing configuration. Aerodynamic data was obtained and used to perform trim analysis, short period oscillation and phugoid analyses. The results indicate that the box wing may have unacceptable handling characteristics.

CAD Implementation

The CAD implementation of the optimized configurations were done in Catia⁸⁹. Catia is a multi-platform CAD/CAM/CAE commercial software suite written in C++ programming language. In building the aircraft models in Catia the aircraft was divided into the following major components:

1. Fuselage.
2. Wings.
3. Engines.
4. Tailplane.
5. Fins.
6. Nose gear.
7. Main gear.

9.1 Procedure - Implementation of Optimized Configurations

The procedure for the implementation of the designs are outlined in the instruction manual of Catia⁸⁹. The appropriate plane for the component design was first chosen then the drawings were done in the sketcher environment. Thereafter, the component was either made solid using the tools in the part design environment or surfaces attached using tools in the generative shape design environment. Where necessary geometric sets, like airfoils coordinates, were imported into the Catia environment. All drawings were done consistent with the results of the optimization performed in Chapter 7.

9.1.1 Conventional Cantilever Wing Aircraft Components

The following diagrams show the screen shots of the Catia component drawing for the major parts of the conventional cantilever aircraft.

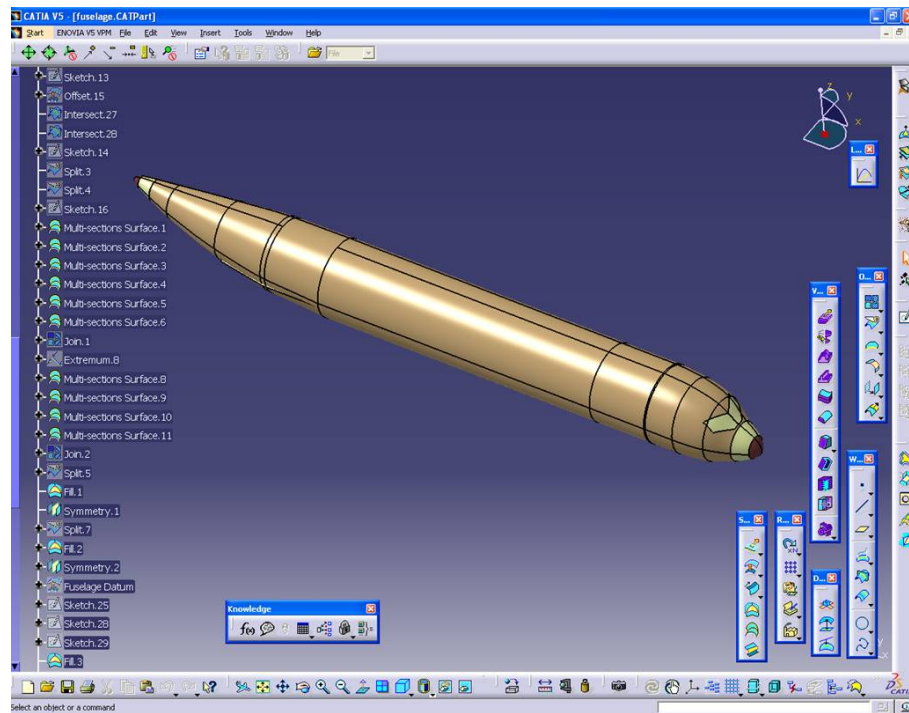


Figure 9.1: Fuselage Screen shot

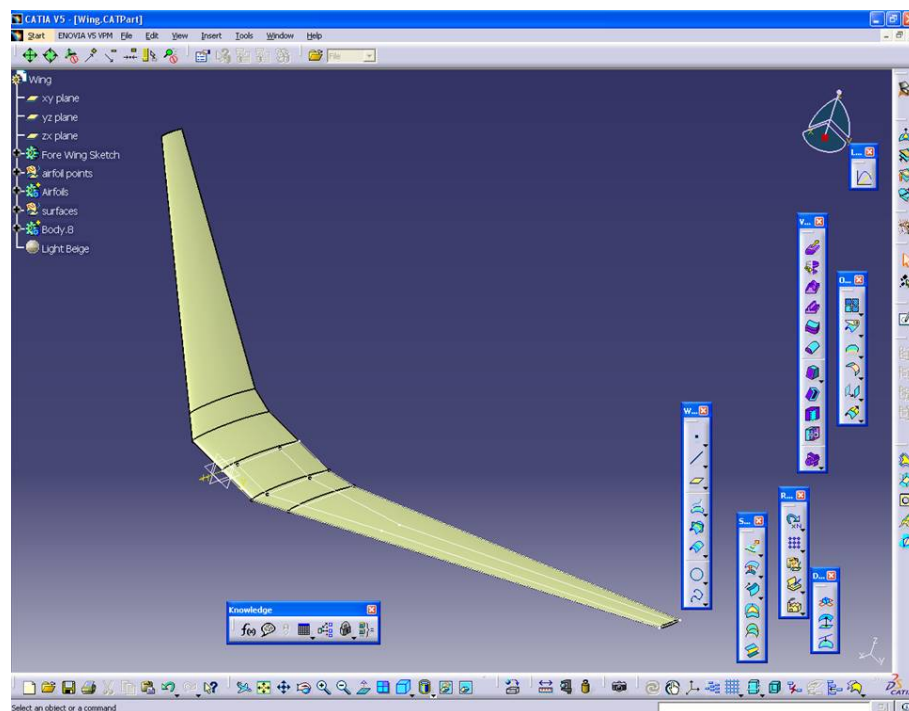


Figure 9.2: Wing Screen shot

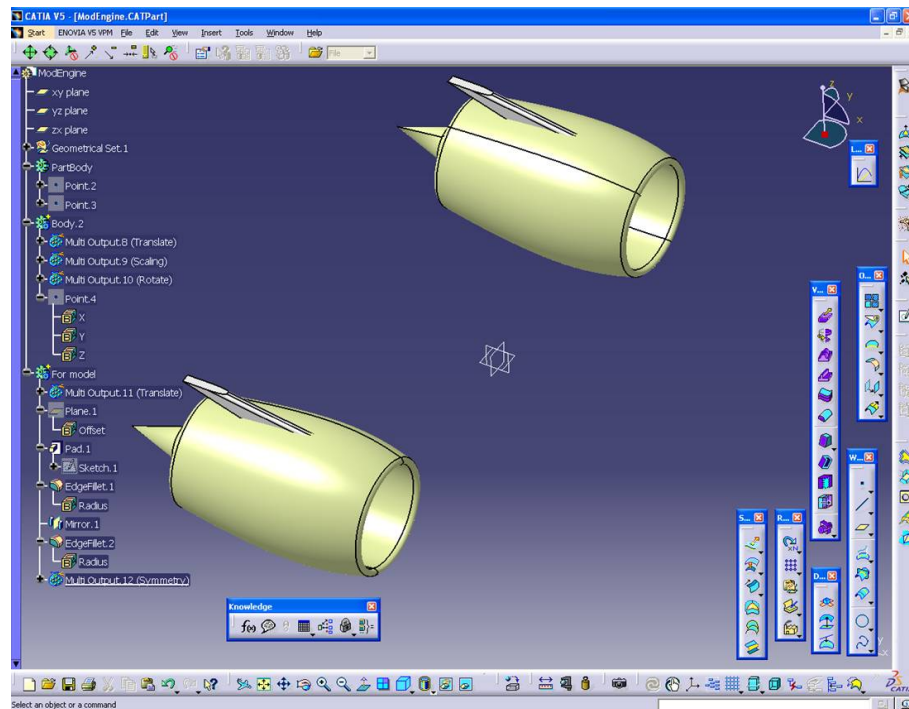


Figure 9.3: Engines Screen shot

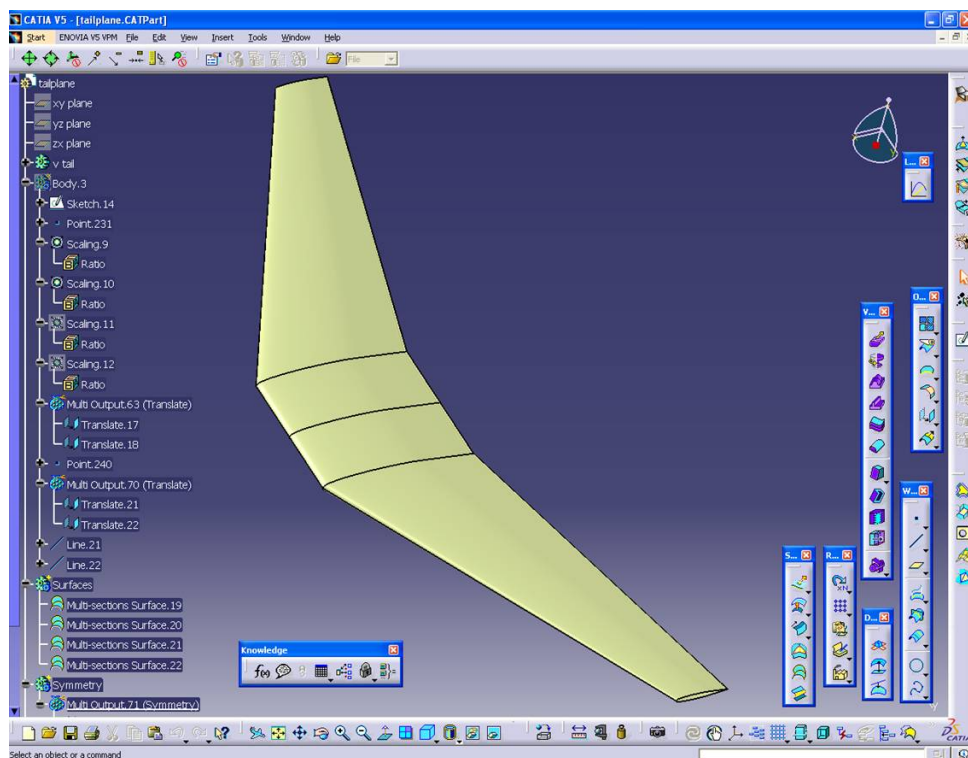


Figure 9.4: Tailplane Screen shot

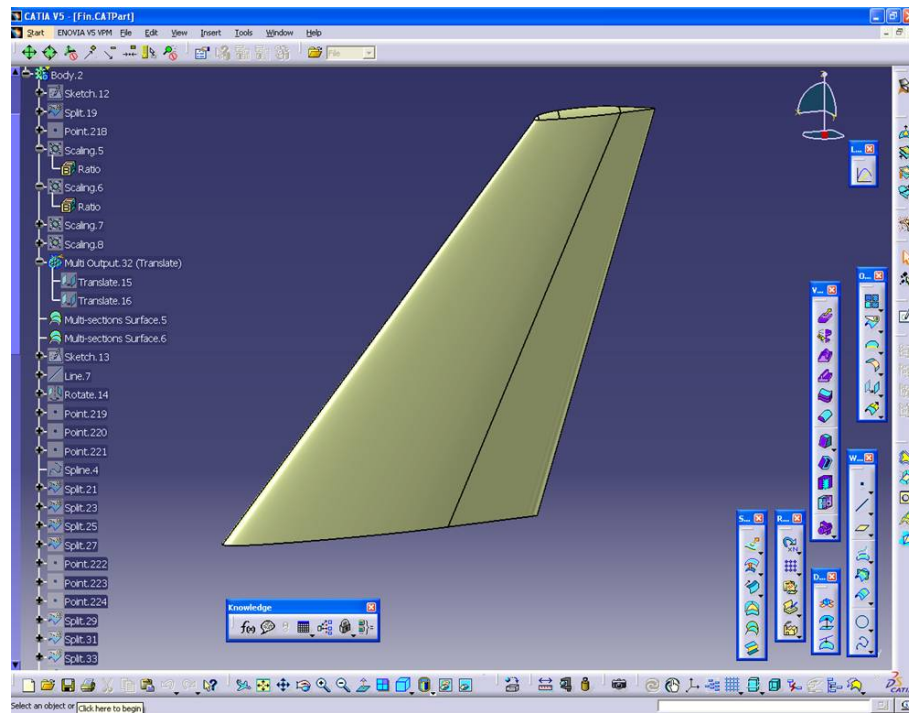


Figure 9.5: Fin Screen shot

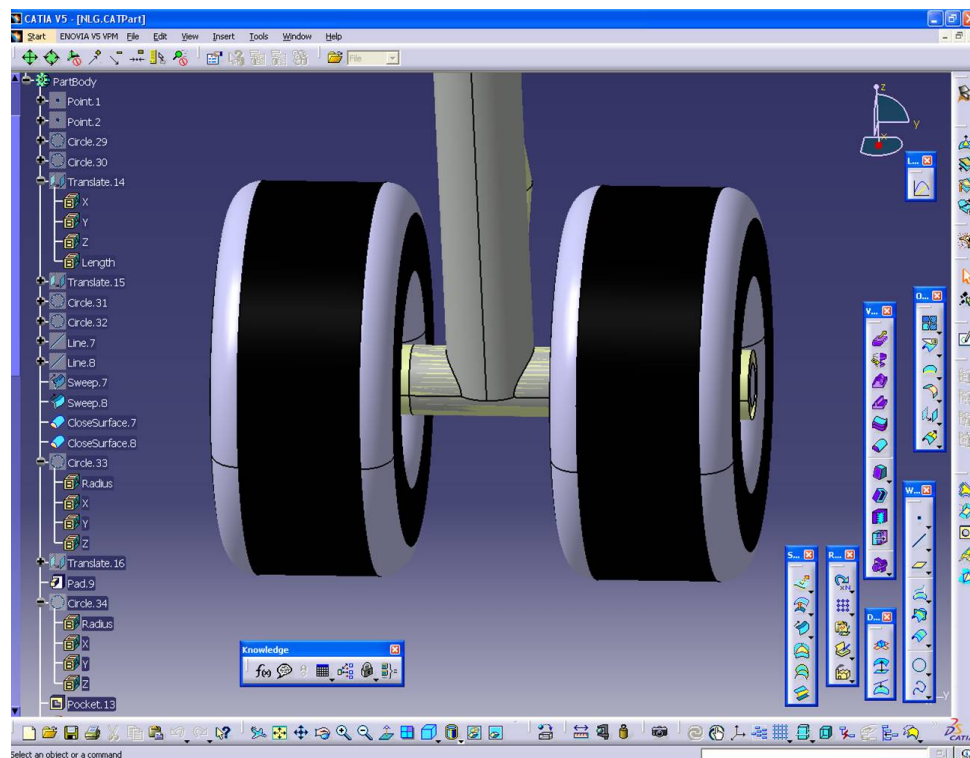


Figure 9.6: Nose Gear Screen shot

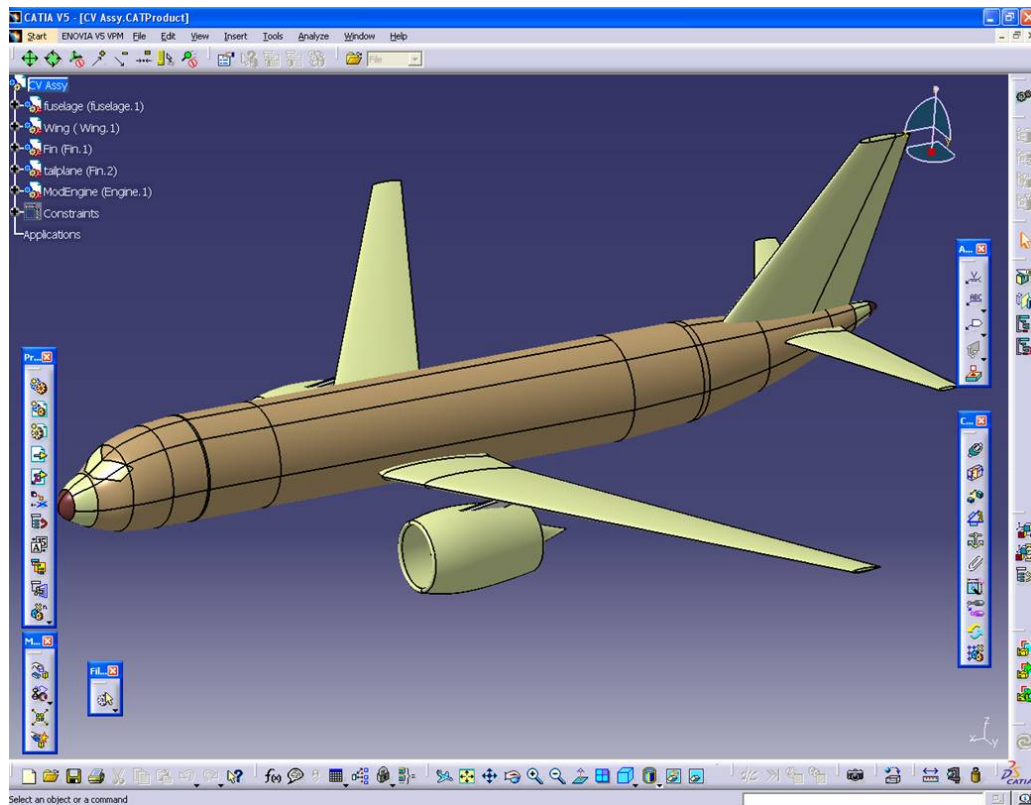


Figure 9.7: Assembled Components Screen shot



Figure 9.8: Rendered Assembled Components Screen shot

9.1.2 Box Wing Aircraft Components

The following diagrams represent the Catia component drawing for the major parts of the box wing aircraft.

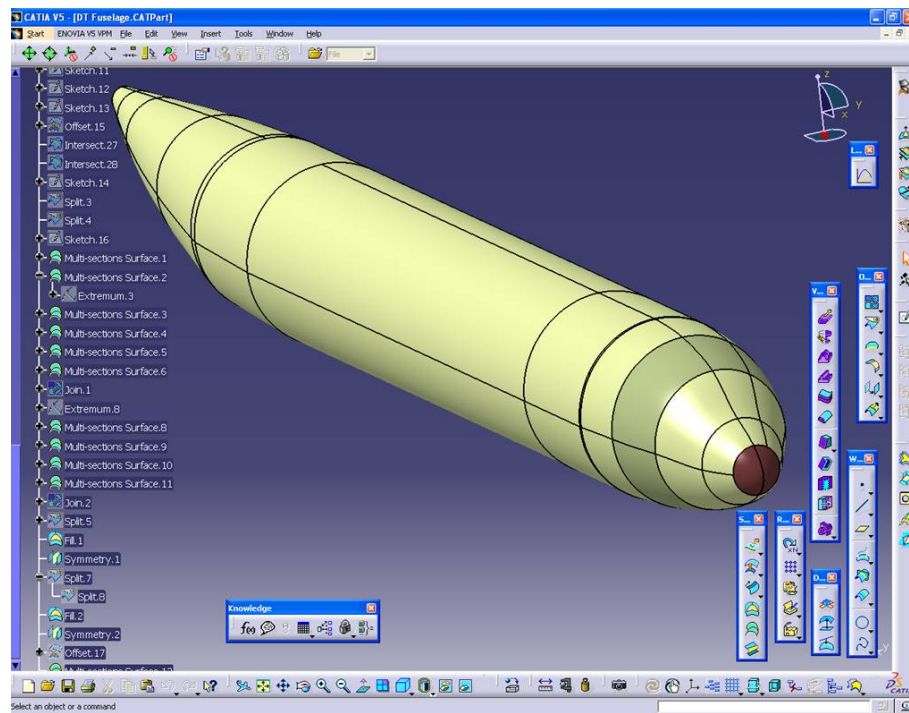


Figure 9.9: Fuselage Screen shot

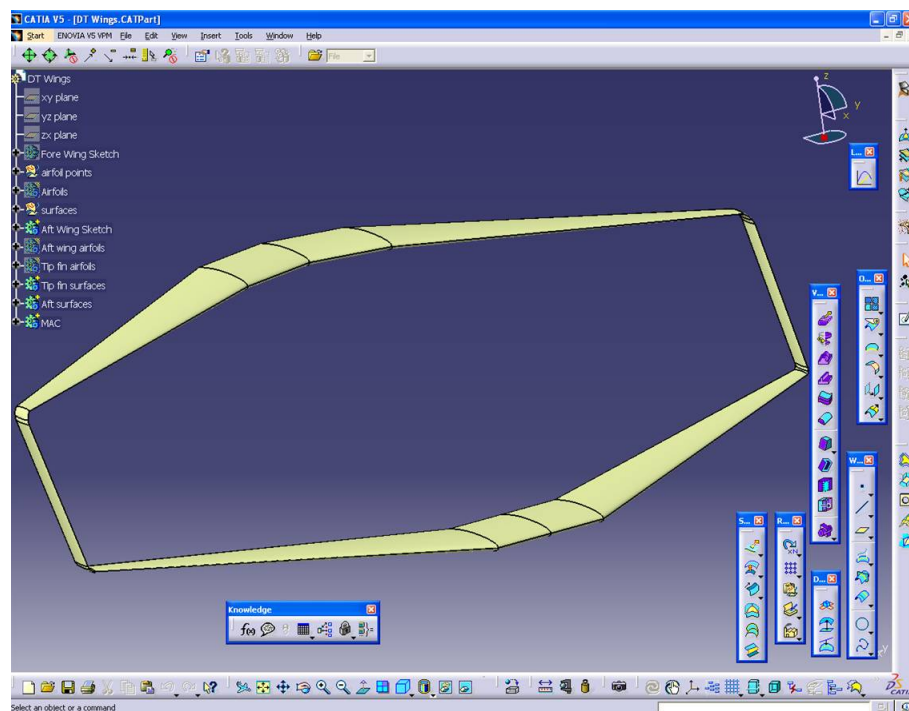


Figure 9.10: Wing Screen shot

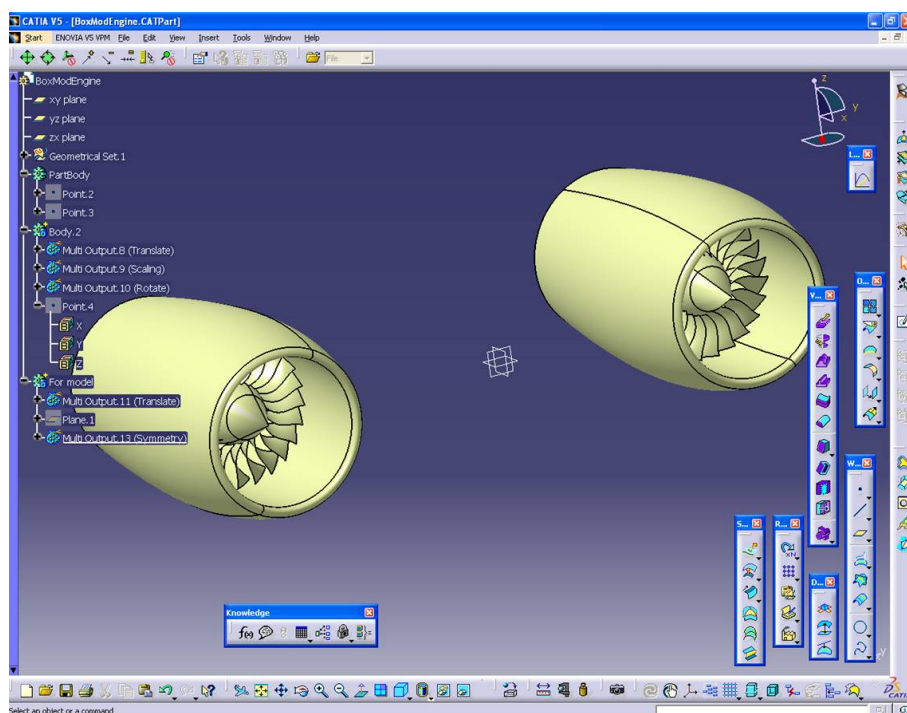


Figure 9.11: Engines Screen shot

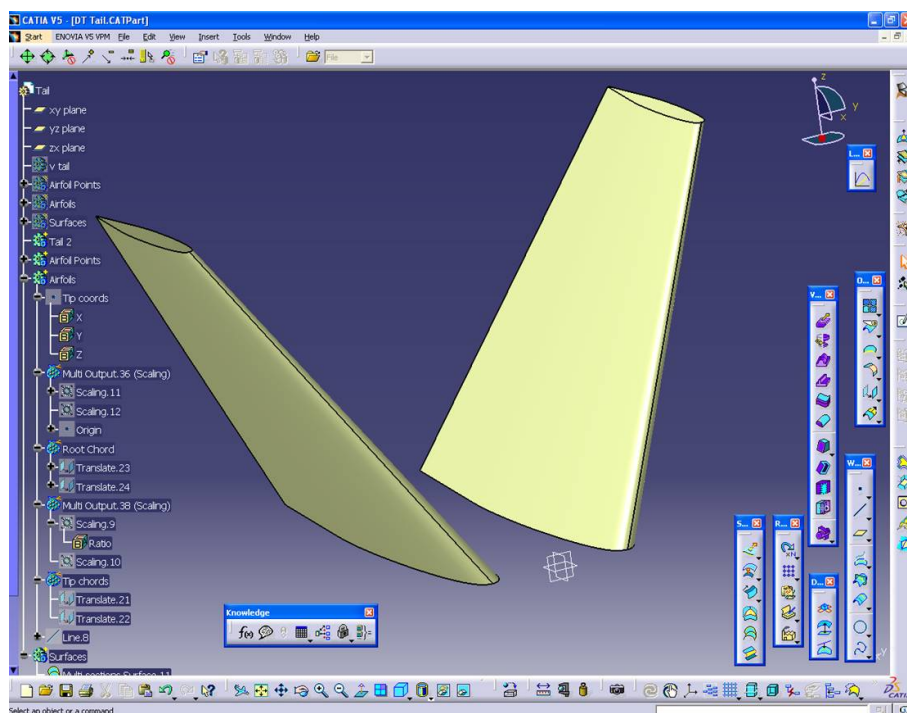


Figure 9.12: Fin Screen shot

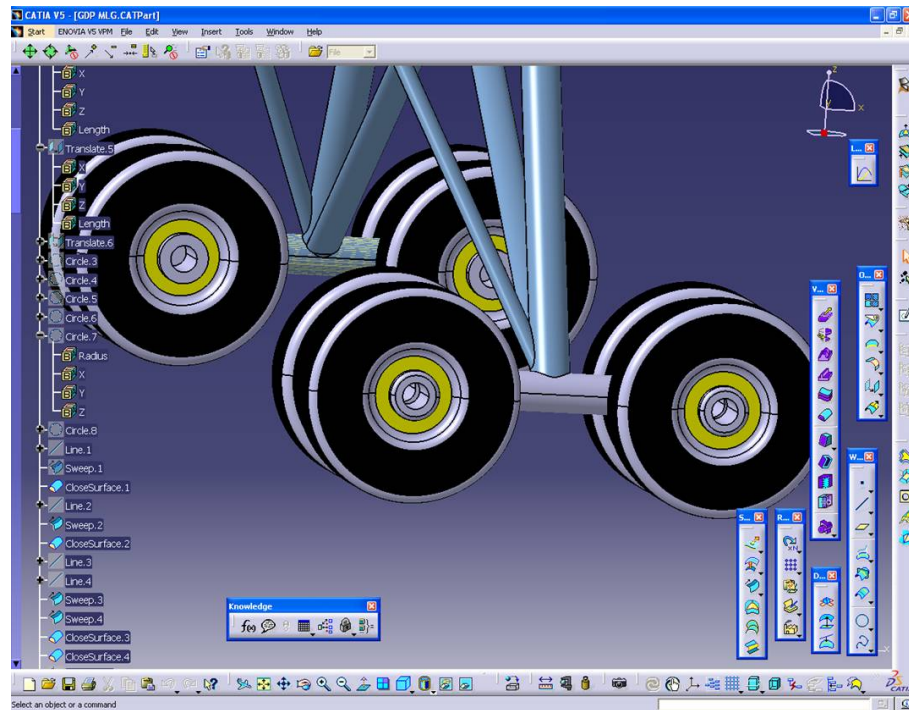


Figure 9.13: Main Gear Screen shot

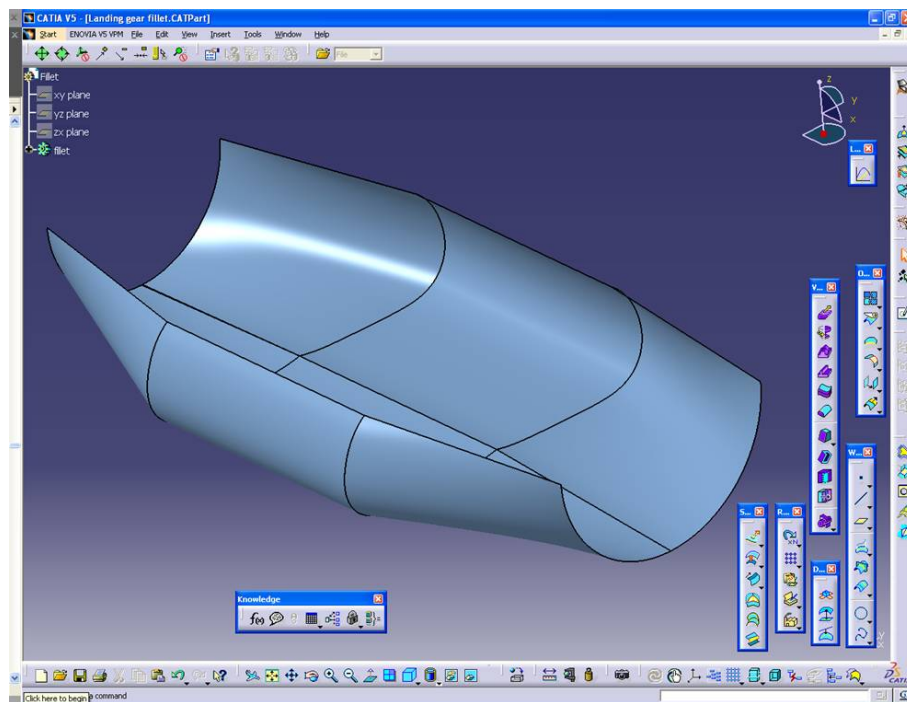


Figure 9.14: Main Landing Gear Fillet Screen shot

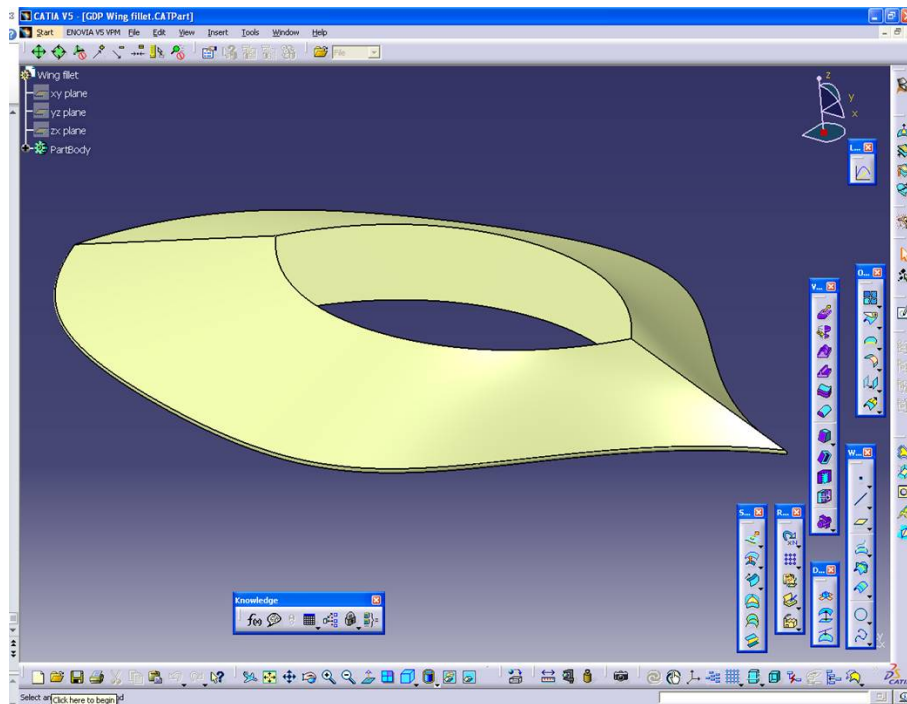


Figure 9.15: Fore Wing Fillet Screen shot

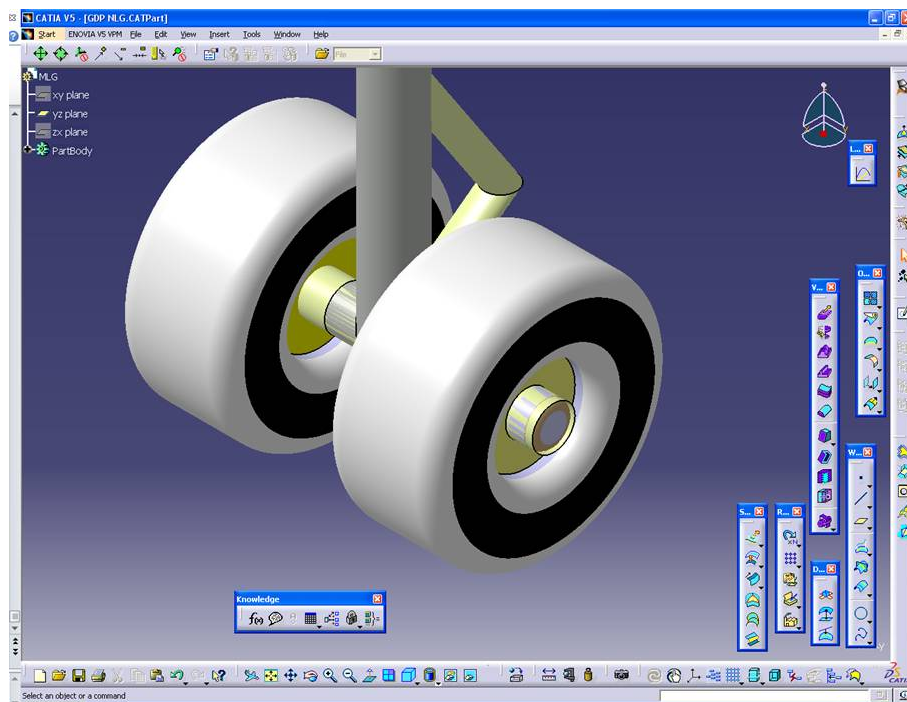


Figure 9.16: Assembled Components Screen shot

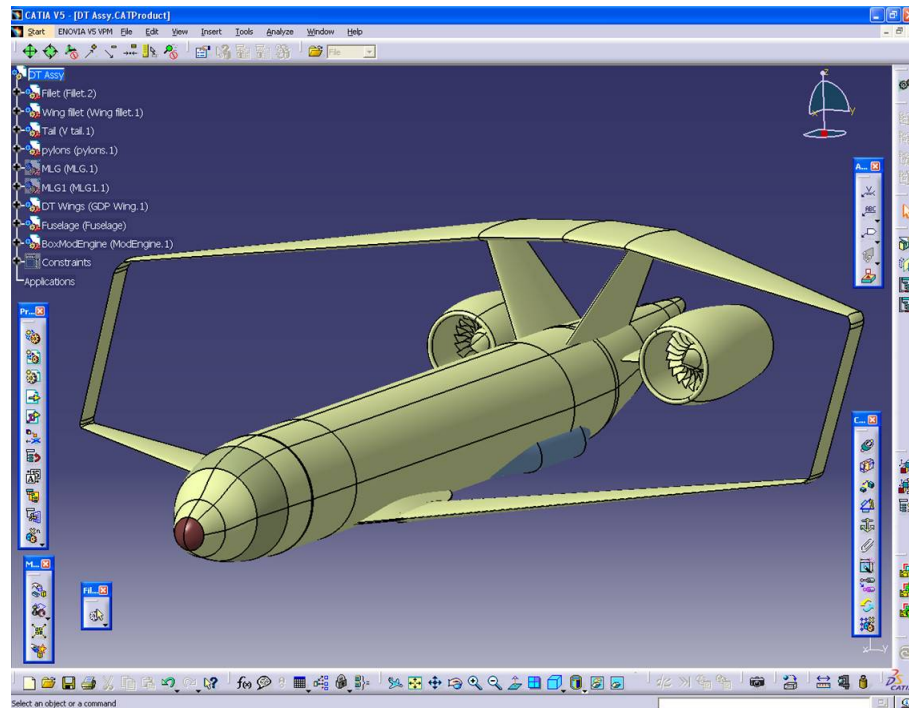


Figure 9.17: Assembled Components Screen shot



Figure 9.18: Rendered Assembled Components Screen shot

Discussion

10.1 Wing Mass Prediction

An aircraft's mass needs to be reasonably estimated at the conceptual design stage for reasons including the relationship between an aircraft's mass, the aircraft's market price and its direct operating cost. This relationship is even more paramount for novel designs where conventional mass estimation methods may not apply. The C_1 (Equation 4.1.5) method used to derive a wing mass prediction method for the box wing was chosen for its adaptability, simplicity and because it includes all the major variable parameters which influence wing mass. In Equation 4.1.5 the coefficient C_1 accounts for the sort of aircraft that is being analyzed, such as long range, short range, braced wing, etc (see Table 4.1).

Equation 4.1.5 was originally developed for aft swept and not forward swept wings. For forward swept wings aeroelastic instabilities such as divergence and flutter become prominent and suggest heavy wings or reduced divergence speed. However, because in the box wing aircraft the aft (forward swept) wing is braced by the fore (aft swept) wing, these aeroelastic instabilities of the aft wings should be mitigated (this is an area for future research). Thus, for conceptual level box wing aircraft wing mass estimations C_1 of Equation 4.1.5 can be appropriately modified.

To obtain the box wing aircraft wing mass correction coefficient, C_1 , a relationship was established between the wing mass of the box wing models and the conventional cantilever wing models. The simplification of idealizing the box wing torsion box cross-sectional geometry into a rectangle avoided the complication of Wolkovitch's⁹ recommendation of having the '*bending-resistant material*' concentrated near the upper leading edges and lower trailing edges. As the analysis was performed assuming an all metal construction this implies that with the use of composite materials the wing masses would be lower.

The wing mass estimation coefficient of 0.028 derived for the fore wing was the same derived for the aft wing and is also the same as that for conventional long range aircraft, see Table 4.1. The result of both wings having the same mass estimation coefficient would be

traceable to the wing aerodynamic loads which are the same on both wings. Furthermore, both wings are connected at the tips by tip fins making the configuration self bracing as each wing would provide some form of load alleviation for the other. In addition, the same set of constraints were applied to both wings during the finite element modeling and this would be another reason for the coefficients to be the same.

The aircraft studied was of medium range for which the C_1 value might be expected to be between 0.028 and the short range value of 0.034. Having the same wing mass prediction coefficient for fore and aft wings makes for a simplification of the design process for box wing aircraft and means that the aft wing would be lighter than the fore wing. This is because typically the aft wing would have a lower sweep angle than the fore wing. This general result is of significance to the conceptual designer, as the difference in fore and aft wing masses would be of influence in the positioning of other heavy items such as engines and landing gears. This result would also be of consideration for center of gravity and static stability issues of the box wing configuration. If allowance is made for aeroelasticity the C_1 value could be less than 0.028.

10.2 Wing/Tip Fin Joint Fixity

In order to determine the best joint for the box wing configuration, it was necessary to outline the criteria the chosen joint should satisfy. Thus, it is essential that the joint produces a low overall wing mass; generally lower wing root bending moment suggests lower wing mass. Also, the joint should not in any way accentuate any potential aeroelastic problems. It is desirable that the best joint produces greater overall wing stiffness. Since the main merit of the box wing configuration comes from its ‘box’ nature, it is desirable that the best joint does not allow any structural behaviour that would perturb the aerodynamics of the wing tip/tip fin junction, which normally requires special design to maximize the benefits of the configuration. Factors such as flutter and divergence should normally be investigated and compared with strength studies to arrive at the most optimal wing joint fixity for final design considerations, but these are beyond the scope of this work.

Table 10.1: Normalized Wing Root Parameters - Fore Wing

Output	Rigid	Universal	Ball	Pin	Cantilever Wing
Out-of-plane BM	1	1.28	1.26	1.26	1.37
Dragwise BM	1	0.21	0.04	0.53	0.00
Out-of-plane SF	1	1.03	1.02	1.02	1.05
Torsion	1	0.02	0.13	0.43	1.02
Tip Deflection	1	1.64	1.62	1.52	1.81

It would appear from Tables 10.1 and 10.2 (reproduced from Section 5.7) that the ball joint would be the best joint for the aircraft. It has the lowest drag-wise bending mo-

Table 10.2: Normalized Wing Root Parameters - Aft Wing

Output	Rigid	Universal	Ball	Pin
Out-of-plane BM	1	1.29	1.31	1.23
Dragwise BM	1	0.39	0.07	0.24
Out-of-plane SF	1	1.03	1.03	1.01
Torsion	1	0.37	0.31	0.63
Tip Deflection	1	1.64	1.68	1.54

ment and comparable shear force. It also has a relatively lower torsional force at the wing root which, overall, should make for a light wing. However, the ball joint permits pitching about the flexural axis and about the local stream-wise Y axis at the wing tip. This freedom to pitch as air loads determine is a drawback. Bending and torsion forces contribute to the twist angle of airfoil sections and are accentuated by aileron deflection. This could lead to decreased aileron effectiveness and even aileron reversal. In addition, the ball joint would interfere in the aerodynamics of the wing tip/tip fin junction as it makes for a more flexible wing structure. Hence, the ball joint is unsuitable for the box wing configuration.

Even though the universal joint may produce a stiffer wing than the ball joint, its higher drag-wise bending moment, out-of-plane bending moment and shear force would make for a heavier structure. Furthermore, the universal joint's freedom to pitch about the local stream-wise axis would distort the aerodynamics of the wing tip/tip fin junction. The universal joint is therefore unsuitable for the box wing configuration.

The pin joint's drag-wise bending moment and torsional force, on both wings, are considerably lower than that of the rigid joint wing. However, its higher wing root bending moment cancels this advantage over the rigid joint. In addition, the joint does not make for the stiffest structure. Furthermore, the movement of the wing tips with respect to the tip fin under load would compromise the aerodynamic benefits of the box wing configuration. The pin joint is therefore unsuitable for the box wing configuration.

From the observed trends, the rigid joint should offer a lighter wing structure than any of the other joints, with its significantly lower wing root out-of-plane bending moment. It is also evident that chord-wise bending moments would be of consideration in sizing the torsion box for wings with this type of joint. However, the joint would not accentuate aeroelastic problems and because it transmits all the stresses (see Figs 5.13 and 5.14), it should produce a heavier tip fin meaning greater inertia relief. However, this increase in the moment of inertia has a consequence of reduced roll responsiveness; an undesirable development for military aircraft but less critical for civil transports. The rigid joint also produces greater overall wing stiffness which could have ameliorating effects on the reduced roll responsiveness caused by a heavy tip fin. Finally, the rigid joint allows for the design of the wing tip/tip fin junction to take full advantage of the aerodynamic

benefits of the configuration. Consequently, the rigid joint is most suitable for the box wing configuration.

10.3 Tip Fin Inclination

Howe⁶³ states that out-of-plane bending moment and shear force are critical to estimating the mass of aircraft wings as they determine the effective end load material for spar web and distributed flanges respectively of the primary wing structural box. Also Figs 10.1 and 10.2 (reproduced from Section 6.7) show that there is little difference in these critical sizing parameters for all the tip fin inclinations. Therefore it is not surprising that there are only minor differences in the torsion box masses of the various tip fins.

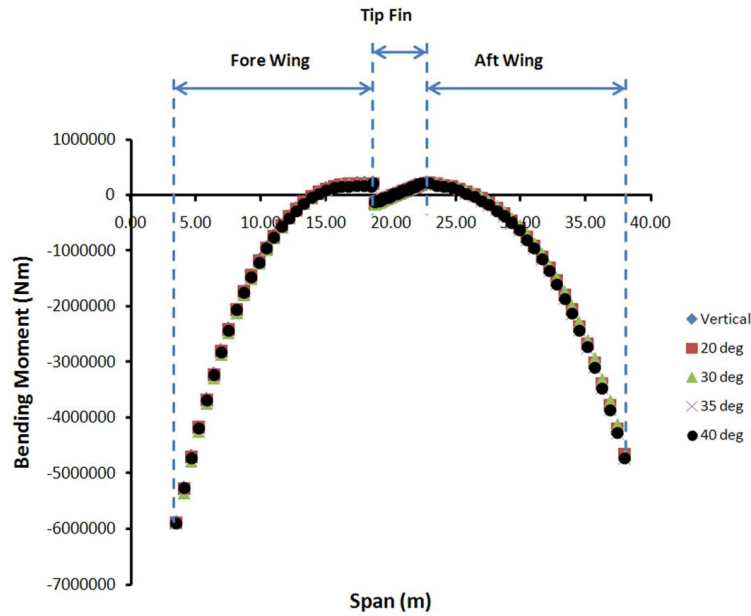


Figure 10.1: Bending Moment Distribution

An examination of this absence of any significant difference due to tip fin inclination begins by considering the stresses in the tip fins. Several factors contribute to this behaviour but five main contributors are identified as bending moments, shear forces (YX and XY), axial loads, torsion (due to airfoil pitching moment and sweep angle mismatch).

The out-of-plane bending moment does not change with change in tip fin angle because it acts in the global Z direction. Similarly, the out-of-plane shear force also acts in the global Z direction and so does not necessarily change with tip fin angle. Torsion changes with tip fin angle as shown in Fig 6.6 but it is not critical to the sizing of any component of the torsion box. Furthermore, the shear force in the XY plane changes with tip fin inclination as shown in Fig 6.8 but again it is not critical in sizing the torsion box.

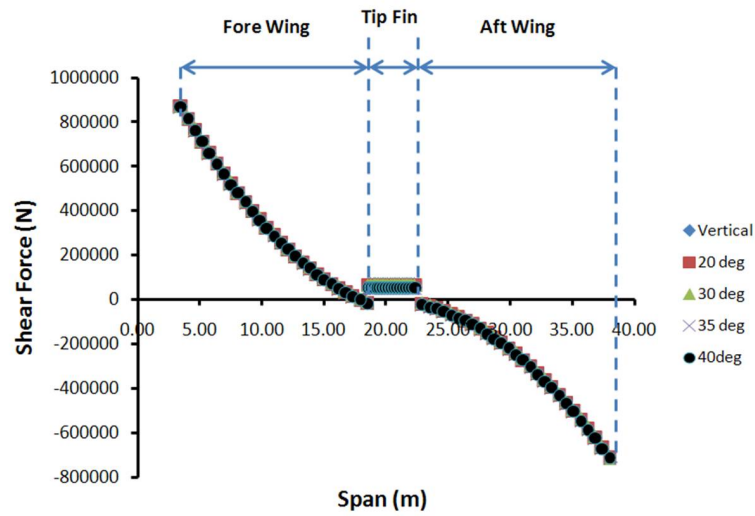


Figure 10.2: Shear Force Distribution

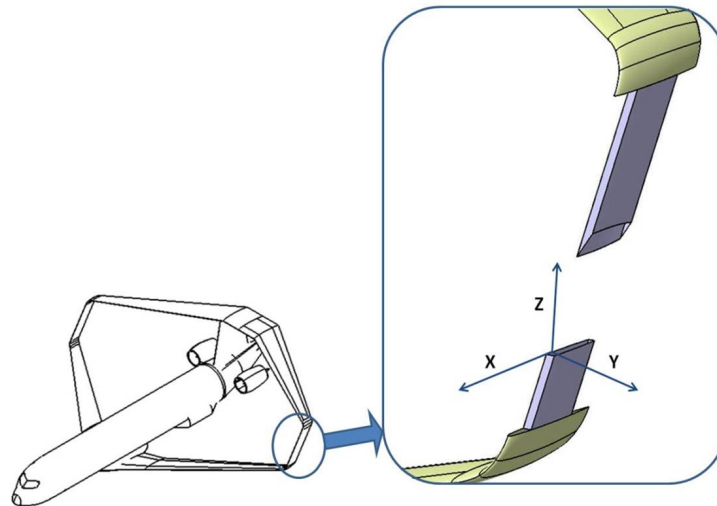


Figure 10.3: Tip Fin Section

Of the five main contributors to the stresses in the tip fin, three remain essentially the same with change in tip fin inclination. The only two which change, Figs 6.6 and 6.8, do not show this because the stiffness of the section is high enough along the moment of inertia not to provoke an appreciable difference in deformation of the tip fin, see Fig 10.3. However, these torsion and dragwise shear force distributions would have significant influences on the wing/tip fin joint design and suggests heavy joints; an area not covered in this research.

Thus, the minor variation in the out-of-plane bending moment, shear force distributions and torsion box masses (Table 10.3) suggests that tip fin inclination has a reduced effect on the structural design of a box wing aircraft. This deduction is valid from a structural

Table 10.3: Normalized Wing Parameters

Tip Fin Inclination	Vertical	20 deg	30 deg	35 deg	40 deg
Wing Tip Deflection	1.00	1.03	1.05	1.06	1.06
Tip Fin Torsion Box Mass	1.00	1.00	1.01	1.04	1.23
Wing Torsion Box Mass	1.00	1.00	1.00	1.00	1.02

viewpoint but tip fin inclination could have non negligible effects on the dynamic modes, flutter speed and frequency; areas not investigated.

10.4 Case Studies

The optimized average thickness to chord ratio for the box wing aircraft of 9% is lower than that of the conventional equivalent aircraft of this class, the B767-200 which has an average thickness to chord ratio of 11.5%⁹⁰. This result is consistent with Wolkovitch⁹ who posits that because the effective beam depth, d , of a joined/box wing is primarily determined by the chord of its airfoils as sketched in Fig 2.2, their thickness is secondary making joined/box wings suitable for thin airfoils. However, as evident in the optimization performed, it was factors like drag, aircraft all up mass and direct operating costs that were used to determine the average thickness to chord ratio and not the effective beam depth of the box wing. For the conventional cantilever wing aircraft its optimized average thickness to chord ratio of 11% is close to that of the B767-200 and also close that of another similar sized airliner the A310-300, whose average thickness to chord ratio is 11.8%⁹⁰. The proximity of these results provides some validation to the optimization methodology.

The wing areas for the B767-200 and A310-300 are $283.30m^2$ and $219.00m^2$ respectively and the optimized result for the box wing aircraft was an area of $224.00m^2$ from the combined fore and aft wing areas of $112.00m^2$ each. Thus, there does not seem to be any correlation with these airliners as far as wing area is concerned. For the conventional aircraft its optimized wing area was $194.00m^2$ which is significantly lower than that of the airliners. This value is in harmony with the current trend in industry of lower wing area for higher cruise lift coefficient. However, an interesting observation is that the total wing area of $224.00m^2$ for the box wing aircraft is equivalent to the sum of the conventional cantilever aircraft's optimized wing area of $194.00m^2$ and its tailplane area, $30.00m^2$. The conventional cantilever wing aircraft's tailplane was initially sized to an area of $72.02m^2$ using methods given in Roskam⁹¹ but later adjusted to $30.00m^2$ in order to satisfy static and dynamic stability requirements. This could mean that for the conceptual design of a box wing aircraft what may be required is to determine the conventional equivalent and divide the sum of the wing and tailplane areas by 2 for fore and aft wings. This is similar to what was done ab-initio by Schiktanz⁹² in his conceptual design of medium range box

wing aircraft. Schiktanz⁹² choose the A320 as the conventional equivalent to compare his design with and he basically divided the A320's wing area into 2 equal areas for the fore and aft wings.

10.5 Stability and Control

A fully representative equation to calculate the neutral point of an aircraft is difficult to develop, but simple equations have been developed for conventional cantilever, canard and even 3 winged aircraft but these are not applicable to box wing aircraft. This is due primarily to the principle of operation of box wing aircraft that requires both wings to produce equal lift forces. However, a simple approximation to the pitching moment equation, which is sufficiently representative for preliminary studies, and which provides considerable insight into the basic requirements for box wing aircraft static stability and trim was developed.

The box wing aircraft principle of operation suggests neutral stability therefore a 2% static margin was imposed to attain marginal intrinsic stability. This meant the fore wing had to generate 2% more lift and the aft wing 2% less lift. This in turn caused a fractional deviation from Prandtl's ideal configuration. Furthermore, as evident in Fig 6.2, there is second reason why the aircraft in this study differs from Prandtl's ideal; the lift distribution on the fore and aft wings are similar but not identical. However, these two reasons together can account for a 0.01% increase in the overall induced drag of the vehicle which is close enough to the ideal not to defeat the purpose of the box wing aircraft. This marginal increase in induced drag is the penalty to pay for intrinsic longitudinal stability.

The conventional aircraft had its neutral point at 54% (see Fig 8.2) of the MAC while the box wing aircraft's neutral point was at 14% (see Fig 8.5) of the virtual MAC. The virtual MAC here being the sum of the MACs of the fore and aft wing MACs. Typically the '*space*' in front of the neutral point on the MAC indicates how much room there is for the cg to travel within the MAC. Thus, the conventional aircraft by reason of its neutral point location being at 54% MAC has more '*room*' for the movement of the cg than the box wing aircraft. This greater latitude for cg movement is consistent with conventional cantilever wing designs⁷⁰ but the box wing requires little movement of the cg hence the neutral point being at 14% MAC is not a drawback in that sense. To ensure that the box wing's cg range remains within limits means such as fuel redistribution can be used as the aircraft has extra fuel capacity in the fins. The box wing's cg range also demands that the nose gear bears loads of about 14% of the aircraft mass, which is high as given in Howe⁵³. Howe states that the nose should bear loads ranging from 6% to 15% at OEM and MTOM respectively. However, for the box wing this load stays virtually constant throughout its limited cg range and so may not be critical.

An interesting observation of the inertia statements of both aircraft is that for the conventional aircraft:

$$I_{xx} + I_{yy} \approx I_{zz} \quad (10.5.1)$$

which is as stated in Bruhn⁸³.

For the box wing aircraft:

$$I_{xx} + I_{yy} > I_{zz} \quad (10.5.2)$$

This incongruence of the two equations could be attributable to the configuration. However, the effect of the I_{zz} on the directional stability of the box wing was not investigated in this study.

Longitudinal trim typically involves the simultaneous adjustment of elevator angle and thrust to give the required airspeed and flight path angle for a given airframe configuration. Equilibrium is achievable only if the aircraft is longitudinally stable and the control actions to trim depend on the degree of longitudinal static stability. Since the longitudinal flight condition is continuously variable it is important that trimmed equilibrium is possible at all conditions. The trimming envelope of the conventional cantilever wing and box wing aircraft are shown in Figs 8.8 and 8.12. Evidently, the trimming envelope of the box wing was smaller than that of the conventional cantilever wing aircraft. This smaller envelope of the box wing is a demerit and restricts the operational versatility of the aircraft, with safety implications. Further investigations using CFD may identify what aspects of the configuration require redesign or modification in order to expand this envelope.

However, it is interesting to compare both aircraft flying at 31,000ft at Mach 0.8. From Table 10.4 both aircraft were cruising at about the same angle of attack but while the conventional aircraft's wing had a positive angle of attack, the box wing's fore wing had a negative angle of attack. At the tailplane and aft wing both had positive angle of attacks.

Table 10.4: Aircraft Parameters at Mach 0.8 31,000ft

Type	AoA	Wing/Fore wing AoA	Elevon	Tailplane/Aft wing AoA	Elevator
	(°)	(°)	(°)	(°)	(°)
Conventional	1.70	2.94		1.12	-0.22
Box	1.68	-1.32	3.1	2.10	-5.13

The fact that for the box wing aircraft the fore wing is at a '*low*' angle and the aft wing a '*high*' angle is line with Bell's¹⁵ highlight that the rear wing induces an upwash on the forward wing, which in turn induces a downwash on the rear wing. Thus, the fore wing's '*low*'

angle of attack is to compensate for the increased angle of attack caused by the upwash on it induced by the aft wing. Similarly, the aft wing's '*high*' angle of attack is to compensate for the reduced angle of attack induced on it by the downwash from the fore wing.

The trim drag of the conventional aircraft with an elevator angle of -0.22° would be much lower than that of the box wing with elevon and elevator angles of 3.10° and -5.13° respectively. This suggests that further optimization is required for the box wing as the trim drag indicated by this simulation could reduce the advantage this configuration has over the conventional aircraft. However, this can be ameliorated by the using the extra fuel capacity in the fins to trim.

The models of the conventional cantilever wing and box wing aircraft with 33% payload were simulated at an altitude of 31,000ft at Mach 0.8. After a period of 2 seconds a step input lasting 0.02 seconds was inputted by the model's elevator and elevon. The results of the excitation for the box wing and conventional cantilever wing aircraft models were then inputted on the '*thumb print*' criterion is shown in Fig 10.4. The figure illustrates the significant difference between the conventional and the box wing aircraft. Whereas the conventional aircraft falls in the satisfactory area of the '*thumb print*' the box wing aircraft is in the unacceptable area. This is partly due to the static margin the box wing which at 2% makes for marginal longitudinal stability and the rather large wing area of the aft wing. If the static margin were increased to achieve the satisfactory area of the criterion the box wing would depart further from its ideal arrangement and negate some of the benefits of the design. On the other hand, since the 2% static margin imposed on the model achieved little it would be logical to recommend stability augmentation systems for the box wing aircraft configuration as this would retain the benefits of the configuration and achieve the satisfactory area of the '*thumb print*' criterion. However, that would be topic for future research and not for this work.

The phugoid mode was excited for the models by applying a small disturbance to the aircraft in trimmed flight. This was achieved by applying a small step input to the elevator and elevon which caused the aircraft to pitch down initially before climbing. By comparison, the amplitude of the box wing aircraft's altitude oscillation was about 1.2 times that of the conventional aircraft. Fig 8.17 shows that even though the box wing aircraft descended 323 ft below start altitude compared to the conventional aircraft which climbed 724 ft above start altitude (Fig 8.16), the box wing had oscillated much more in amplitude. There is a similar trend when the speed losses in Figs 8.16 and 8.17 are compared. While the conventional aircraft lost 62 kts in the first oscillation, the box wing lost about 3 times that value. This rather high amplitudes of the box wings oscillations are attributable to its marginal stability which would be sensed by any passengers in the aircraft. Again, this provides further substantiation for the need for stability augmentation devices on this configuration of aircraft. However, the overall trends of the graphs are consistent with phugoids.

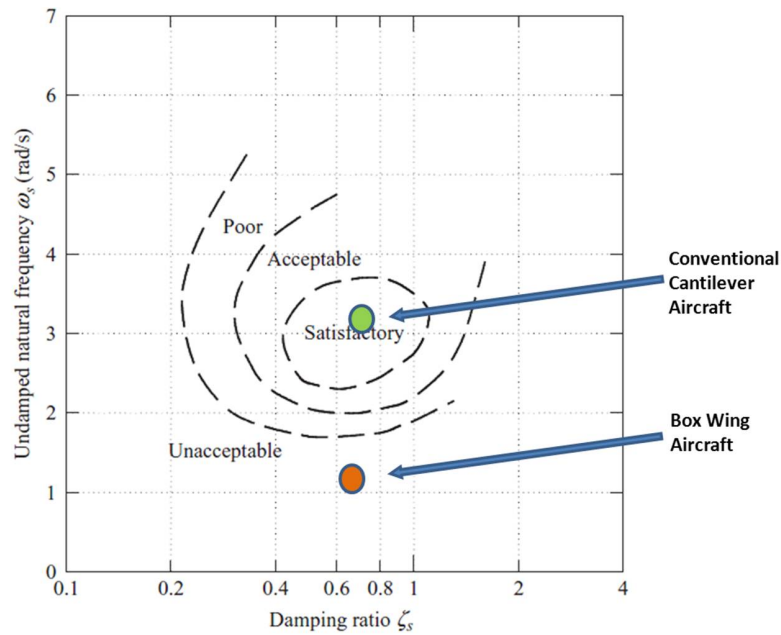


Figure 10.4: Thumb Print Criterion - Box Wing and Conventional Aircraft

10.6 Comparison of Models with In-service Aircraft

The optimum box wing and conventional configurations comparison with the B767-200 and A310-300 are presented in Table 10.5. It needs to be stated that the box wing and conventional cantilever wing models are based on an all metal construction unlike the B767-200 and A310-300 which use some advanced materials in their construction.

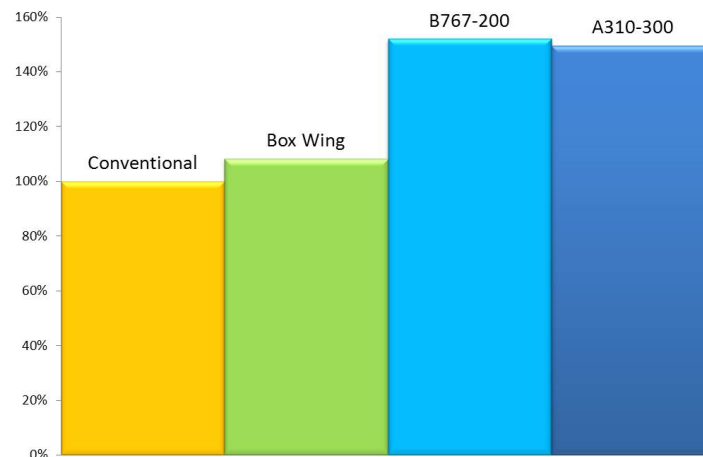


Figure 10.5: OEM Comparisons

From Fig 10.5, the box wing aircraft's OEM came out 8% heavier than the conventional aircraft's OEM. Since both use the 'same' fuselage, the extra mass can be attributed to the structural mass of the box wing tip fins and the attachments of the wing configuration to the fuselage. However, both OEMs are at least 50% lighter than the B767-200's and the A310-300's. The payload of the box wing and conventional aircraft are both 9% less than

Table 10.5: Geometric Weights and Performance Outcomes

Aircraft Type	Optimum Conventional	Optimum Box Wing	B767-200 ⁹⁰	A310-300 ⁹⁰
External Dimensions				
Wing span (m)	47.00	37.6	47.57	43.89
AR - fore/aft	11.39	12.62/12.62	7.99	8.80
S - fore/aft (m^2)	194	112/112	283	219
Average t/c (%)	11	9	11.5	11.8
$\Lambda_{1/4}$ - fore/aft ($^\circ$)	27.0	28.0/-24.0	31.5	28.0
Max length (m)	46.00	46.00	47.24	45.13
Max dia (m)	5.60	5.60	5.03	5.64
Masses				
$OEMs$ (kg)	53250	57605	80921	79666
Design payload (kg)	31050	31050	33912	20710
M_{TOM} (kg)	114916	114240	136078	150000
Max fuel (kg)	46630	37692		
Max pax	270	270	290	280
Performance				
LFL (m)	1783	1615	1463	1490
TFL (m)	1640	1336	1770	2290
Design range (nm)	4000	4000	3220	4300
Market price (2007USD)	108.7m	121.9m		
Fuel/pax/nm (kg)	0.0283	0.0237	0.0305	0.0529
DOC/nm (USD/nm)	20.88	20.34		
DOC/nm/seat	0.077	0.075		

that of the B767's 33,912kg and 30% more than the A310's 20,710kg as shown in Fig 10.6.

**Figure 10.6:** Payload Comparisons

For M_{TOM} , the box wing is a marginal 1% lighter than the conventional aircraft, see Fig 10.7. This is because the conventional aircraft required 30,616kg of fuel to fly the design mission range of 4000nm; the box wing used only 25,585kg. The optimized average thickness to chord ratio of 9% and 3.32 m mean aerodynamic chord meant that the maximum fuel capacity of the box wing at 37,692kg is about 9 tons less than the conventional aircraft's 46,630kg. The conventional aircraft's average thickness to chord ratio of 11% and mean aerodynamic chord of 4.23 m makes for its larger fuel carrying capacity.

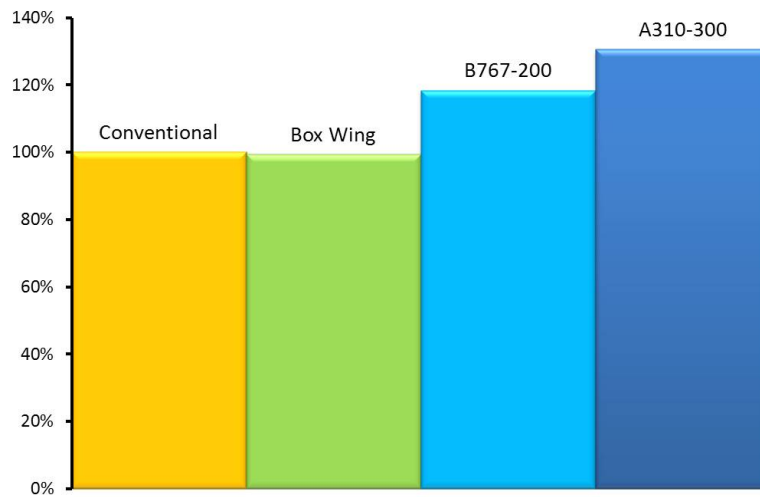


Figure 10.7: M_{TOM} Comparisons

The box wing fuel distribution in percentages is shown in Fig 10.8. The fuel carried in the fore and aft wings add up to 32,650kg which is enough for the design mission. The fins fuel capacity can therefore be available for trimming which would improve the efficiency of the design.

Their fuel capacities notwithstanding, the conventional and box wing M_{TOM} , shown in Fig 10.7, are at least 15% less than the B767's which carries 290 passengers. The same can be said of the A310 which is 31% heavier and carries 10 extra passengers more than the conventional and box wing aircraft.

The overall field performance of the box wing is clearly better than that of the conventional aircraft, see Fig 10.9. Its landing field length is 10% less than the conventional aircraft while its takeoff field length is also 20% shorter. However, the landing field length of the B767-200 is shorter than that of the conventional aircraft and that of the box wing. The B767-200's takeoff field length is longer than that of the box wing and conventional aircraft. For the A310-300, it has a 15% shorter landing field length compared to the conventional aircraft but its takeoff length is 40% longer.

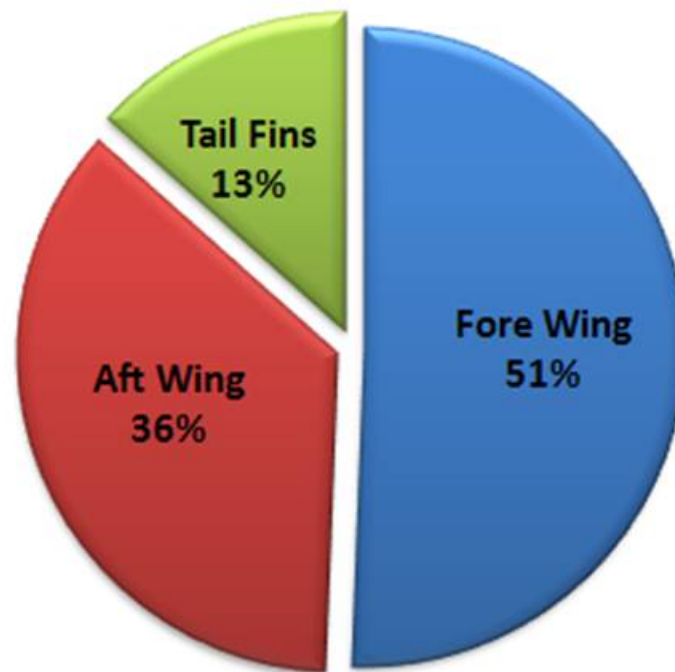


Figure 10.8: Box Wing fuel Distribution

10.7 Box Wing Economic Potential

The predicted market price of the box wing of 121.9m USD is 12% more expensive than the conventional aircraft's 108.7m USD. This is to be expected of a new configuration like the box wing and was accounted for in the cost module of the MDO suite by the '*new programme difficulty factor*' inputted in the algorithm as given in Roskam⁷⁷.

As regards the fuel/pax/nm shown in Fig 10.10, the box wing is about 16% better than the conventional aircraft, 24% better than the B767-200 and about 100% better than the A310-300. The box wing is just 3% better than the conventional aircraft with respect to

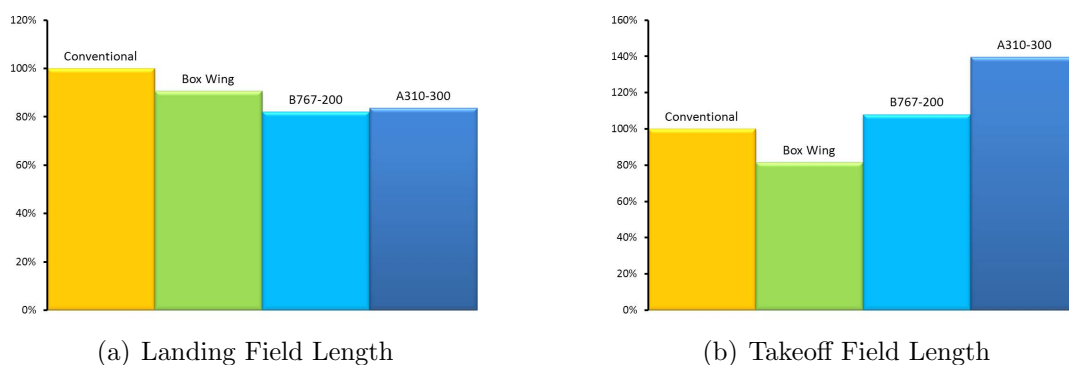


Figure 10.9: Field Performance Comparisons

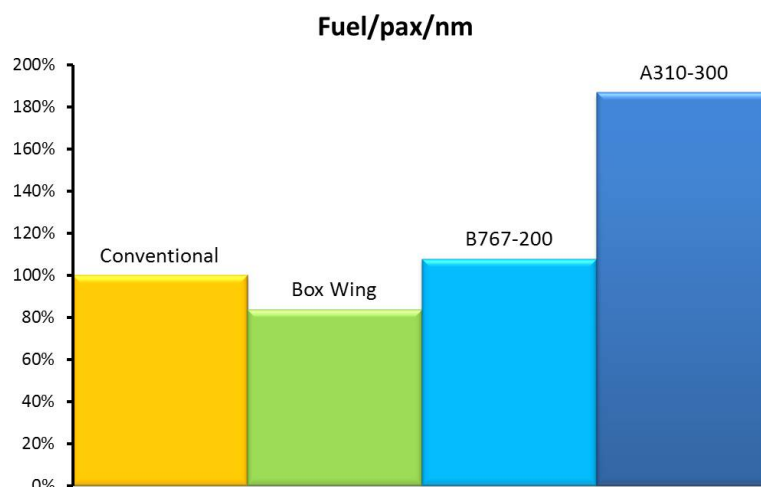


Figure 10.10: Fuel/pax/nm Comparisons

DOC/nm. This does not appear to be much but considering that the computation was based on 2.5 USD per gallon of fuel as it was in 2007 it is instructive to see the trend with increase in fuel price.

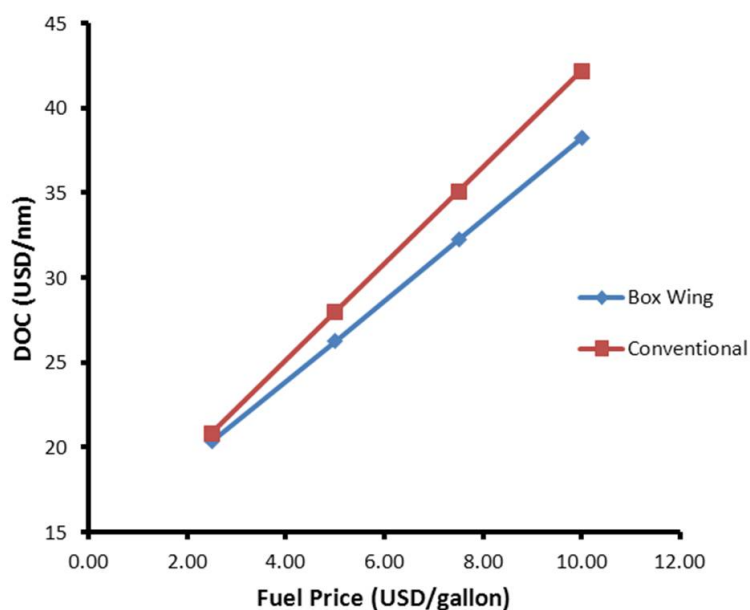


Figure 10.11: DOC/nm Trend with Fuel Price Increase

This is shown in Fig 10.11 which indicates how the box wing compares with the conventional as a result of increase in fuel price. The box wing aircraft's DOC/nm benefit improves from 97% of that of the conventional cantilever wing aircraft at 2.5 USD per gallon to 90% at 10 USD per gallon. Thus, with the likelihood of fuel prices continuously increasing and the carbon tax already introduced in Europe, the box wing has a clear

advantage over conventional designs. In addition, the reduced wing span of the box wing makes it suitable for large long range designs that would easily fit into the required 80m box at airports.

Conclusion and Recommendations

This chapter presents the conclusions of the research. It outlines the inferences from the objectives set out at the beginning of the thesis, the research's contributions to knowledge and the limitations of the research. Recommendations for future work and the author's publications round up the thesis.

11.1 Principal Findings and Research Objectives

The aim of this research was to produce a conceptual design and optimization methodology for box wing transport aircraft. This required examining structural, aerodynamic, and stability and control the issues associated with box wing designs. It also required evaluating economic potential of box wing aircraft designs. This Section highlights the key findings of the research and shows how they were met, in line with the set objectives in Chapter 1.

11.1.1 Wing Mass

This research set out to develop a wing mass estimation correction coefficient, C_1 , for box wing aircraft. The wing mass estimation coefficient of 0.028 derived for the fore wing proved to be the same as that derived for the aft wing. The significance of this is that the aft wing of a medium range box wing aircraft would be lighter than the fore wing because for a medium range box wing aircraft the sweep angle of the aft wing would typically be less than that of the fore wing (wing area being the same), the resulting mass of the aft wing would be lower. This general result is of importance to the conceptual designer, for the difference in mass would be of consideration for center of gravity and static margin issues of the configuration. The mass difference would also be of influence in the positioning of other heavy items such as engines and landing gears. If allowance is made for aeroelasticity the C_1 value could be less than 0.028.

11.1.2 Wing/tip fin joint fixity

It was the objective of this study to investigate box wing aircraft wing/tip fin joint fixity. The rigid joint emerged the preferred wing tip/end fin joint type for the box wing configuration. It produced a stiffer, lighter and more aerodynamically advantageous joint. A rigid joint also produces a loading condition on the rear wing which is less likely to promote stiffness instability in the rear wing.

11.1.3 Tip fin inclination

This research also set out to investigate box wing aircraft tip fin inclination. The minor variation in the out-of-plane bending moment, shear force distributions and torsion box masses suggests that tip fin inclination has a reduced effect on the structural design of a box wing aircraft. This deduction is valid from a structural viewpoint but tip fin inclination may have non negligible effects on the dynamic modes, flutter speed and frequency.

11.1.4 MDO Methodology

This research set out to develop a novel MDO methodology for box wing aircraft. The MDO methodology developed had an integral structural optimization suite and was used to optimize both the conventional cantilever wing and box wing aircraft. The results of the optimization of the conventional cantilever wing aircraft showed approximation with the equivalent in-service aircraft, the B767-200 and therefore provides validation for MDO methodology and for the results of optimization of the box wing aircraft. The wing area (fore and aft wings) of the box wing was equal to the combined areas of the wing and tailplane of the conventional cantilever wing aircraft. Overall the optimized box wing showed to be better than the optimized conventional cantilever wing aircraft by way of field and flight performance.

11.1.5 Longitudinal Stability

It was part of the objective of this study to investigate stability and control aspects of box wing aircraft. The investigations reveal that the box wing aircraft falls in the unacceptable area of the short period oscillation criteria. This is due in part to the principle of operation of the box wing and the fact that its 2% static margin makes for marginal longitudinal stability. To retain the aerodynamic advantages of the box wing configuration stability augmentation devices would be required. Furthermore, the extra fuel capacity in the tail fin can be used to reduce trim drag of the box wing. This is similar to the proposal by Gallman et al⁴¹ concerning the ‘*Wolkovitch Joined Wing*’.

11.1.6 Economics

This research set out to investigate economic aspects of box wing aircraft. At times of low fuel prices the conventional cantilever wing aircraft is better economically than the box wing aircraft. However, the merits of the box wing become apparent with rise in fuel prices and the introduction of carbon taxes. This is in harmony with Gallman et al's⁴³ findings in their optimization of '*Wolkovitch Joined Wing*'. The box wing is therefore potentially the configuration of the future.

11.2 Contributions to Knowledge

The following are the contributions of this research to knowledge:

1. The wing mass estimation correction coefficient of the fore and aft wings of a medium range box wing aircraft when using Howe's⁷² equation is 0.028.
2. The wing mass estimation correction coefficient of the fore and aft wings of a medium range box wing aircraft are the same and therefore the aft wing would almost always be lighter than the fore wing.
3. The rigid joint is the most preferable wing/tip fin joint fixity type for box wing aircraft.
4. Tip fin inclination has minimal effect on the structural design of box wing aircraft.
5. In the conceptual design of a box wing aircraft its wing area can be chosen equal to the sum of wing and tailplane areas of its equivalent conventional cantilever wing aircraft.
6. Box wing aircraft designed based on Prandtl's theory for minimum drag would always require stability augmentation systems to achieve acceptable longitudinal dynamic behaviour.
7. The economic advantages of the box wing over the conventional cantilever wing aircraft improve with increase in fuel price.

11.3 Limitations

Due to the conceptual nature of this research, the analytic models had to be simplified such as the idealization of the wing torsion box cross-sectional geometry into a rectangle. It was also necessary to simplify models in order to reduce data preparation and the turn-around time of data processing.

11.4 Recommendations for Future Work

In view of the foregoing and for more detailed insight to the box wing aircraft it is recommended that:

1. Further studies that account for the tilted bending axis of the box wing configuration be performed for the wing mass estimation coefficient, wing/tip fin joint fixity and tip fin inclination.
2. A non-linear analysis be performed to identify post buckling behaviour of the box wing system.
3. Flutter and divergence analysis of the box wing configuration be performed for a more complete aeroelastic investigation into the effects of joint fixity, tip fin inclination and the effects of the fore and aft wing sweeps on box wing aircraft.
4. Investigations of directional stability and control aspects of box wing aircraft are performed.
5. A CFD analysis be performed on the optimized box wing model for greater understanding of its aerodynamic characteristics.

11.5 Authors Publications

11.5.1 Journals

1. Jemitola PO, Fielding J and Stocking P, (2012) '*Joint Fixity Effect on Structural Design of a Box Wing Aircraft*', published in The Aeronautical Journal, Volume 116, No 1178, April 2012 edition.
2. Jemitola PO, Monterzino G, Fielding J and Lawson C, (2012) '*Tip Fin Inclination Effect on Structural Design of a Box Wing Aircraft*', published online on 5 January 2012 in the Proceedings of the Institution of Mechanical Engineers, Part G: Journal of Aerospace Engineering.
3. Jemitola PO, Fielding J and Monterzino G, (2012) '*Wing Mass Estimation Algorithm for Medium Range Box Wing Aircraft*', accepted for publication in the January 2013 edition of The Aeronautical Journal.

11.5.2 Conference

1. Jemitola PO and Fielding J, (2012) '*Box Wing Aircraft Conceptual Design*', accepted for presentation at the 28th Congress of the International Council of the Aeronautical Sciences (ICAS 2012) to be held in Brisbane, Australia, 23-28 September 2012.

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Visual Basic for Application Macros

A.1 Mass Module VBA Macro One - RangeGSeek

```

Sub RangeGSeek()
Application.ScreenUpdating = False
Application.DisplayStatusBar = True
Application.StatusBar = "Please wait while performing task

finalrow = Cells(Rows.Count, 10).End(xlUp).Row
For i = 18 To finalrow
Do
Cells(i, 24).GoalSeek Goal:=Cells(14, 17), ChangingCell:=Cells(i, 22)
Cells(i, 27).GoalSeek Goal:=0, ChangingCell:=Cells(i, 29)
Loop Until Cells(i, 25).Value2 - 02 < 0.000001
Next i
MsgBox Macro finished
Application.StatusBar = False
Application.ScreenUpdating = True
End Sub

```

A.2 Mass Module VBA Macro Two - SelectAUM

```

Sub SelectAUM()
finalrow = Cells(Rows.Count, 9).End(xlUp).Row
For i = 19 To finalrow
If Cells(i, 9).Value < 29 Then
Cells(i, 31).Value = Cells(i, 29)
Else
Cells(i, 31).Value = OutOfRange
End If
Next i

```

End Sub

A.3 Aerodynamics Module VBA Macro - SelectCDo

```
Sub SelectCDo()  
finalrow = Cells(Rows.Count, 5).End(xlUp).Row  
For i = 29 To finalrow  
If Cells(i, 3).Value < 29 Then  
Cells(i, 15).Value = Cells(i, 13)  
Else  
Cells(i, 15).Value = OutsideRange  
End If  
Next i  
End Sub
```

A.4 Cost Module VBA Macro - AcCost

```
Sub AcCost()  
Dim DOC As Worksheet  
Dim Cost As Worksheet  
Set Cost = Worksheets(Cost)  
Set DOC = Worksheets(DOC)  
finalrow = DOC.Cells(Rows.Count, 2).End(xlUp).Row  
For i = 8 To finalrow  
Cost.Cells(4, 5).Value = DOC.Cells(i, 6) Copies the MTOM to the Cost worksheet  
Cost.Cells(4, 4).Value = DOC.Cells(i, 8) Copies the Airframe weight to the Cost work-  
sheet  
DOC.Cells(i, 16).Value = Cost.Cells(69, 4) Copies new cost of model to the DOC work-  
sheet  
Next i  
End Sub
```

Baseline Aircraft Specifications

B.1 Baseline Medium Range Box Wing Aircraft

The baseline medium range box wing aircraft is taken from Smith and Jemitola⁴³ which is derived from Jemitola⁶¹. It came out of the awareness that the impact of air travel on the environment have been negative and that the current trend is likely to worsen in the future. Directgov⁹³, which publishes information produced by the Central office of Information from UK government departments, states that air travel currently accounts for 6.3 percent of UK total CO_2 emissions. With the change in world climate likely to affect all inhabitants of the world stricter legislations can be expected and therefore novel designs that would offer reduced impact on the environment are being investigated. It is along this approach of new airframe designs with promise of improved fuel efficiency that the baseline box wing design shown in Fig A-1 was performed.

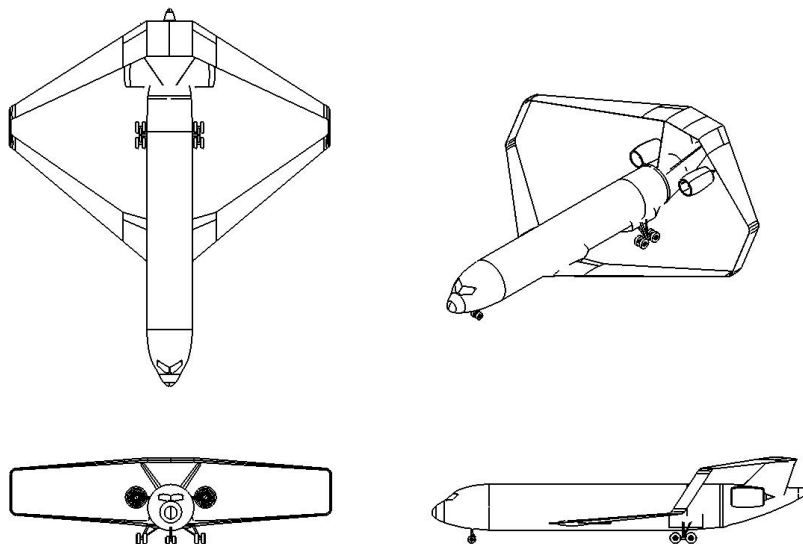


Figure B.1: Baseline Box Wing Aircraft

B.1.1 Design Requirements

The requirements for the box wing aircraft were drawn from work earlier done by Mistry⁹⁴ and the silent aircraft initiative. The general specifications were a 270 single class passenger capacity over a 4000nm range at Mach 0.8. Cruise altitude was specified as 31,000ft while takeoff distance was given as 2500m. By this specification the aircraft falls in the medium range transport category similar to the Boeing 767-200.

B.1.2 Methodology

Unlike conventional aircraft there is generally a relative scarcity of information with regards to design procedures for box wing aircraft. Therefore, time honed conventional aircraft design procedures outlined in Raymer⁷¹ were modified and followed. Information of similar sized conventional aircraft was retrieved and used to estimate the empty mass and fuel mass fractions and subsequently the initial mass statement. This was then used to estimate the engine size and wing areas. A parametric constraint analysis was then performed using methods given in Howe⁵³ and the fuselage geometry defined. The tail fin was sized using methods in Jenkinson⁵⁴ and a more detailed mass estimation subsequently performed. The wing geometry and assembly underwent an elementary parametric optimization process that included airfoil selection. An aerodynamic and performance estimation was then implemented using methods outlined in Jenkinson⁵⁴, Raymer⁷¹ and Roskam⁸⁴. Longitudinal static stability and neutral point determination was done using Philips⁷⁹ as reference and assuming the fore wing to be a canard since there wasn't any simplified specific neutral point determination method for box wing aircraft available in literature. Landing gear details were determined using Raymer⁷¹ while position and loading were chosen consistent with Howe⁵³. Field performance was evaluated using methods given in Raymer⁷¹ and consistent with Ojha⁹⁵, Eshelby⁹⁶, Kermode⁹⁷, Houghton and Carruthers⁹⁸. Cost issues were performed by taking an average of the outcomes of the methods in Raymer⁷¹, Roskam⁷⁷ and Burns⁹⁹. Finally, overall effect on the environment was appraised by comparing the design mission fuel to the product of the maximum passenger capacity and the design range. Other details of the design are in Table A.1.

B.2 Baseline Conventional Cantilever Wing Aircraft

Also performed in Jemitola⁶¹ and to provide a platform for comparison, a design process was undertaken for a conventional cantilever configuration aircraft as shown in Fig A.2. The aircraft was sized same as the box wing to carry 270 passengers and with the same wing area as the total wing area of the box wing aircraft. It was also designed to cruise at the same speed and altitude as the box wing aircraft following the processes outlined in para A.1.2. Details of the aircraft are in Table A.2.

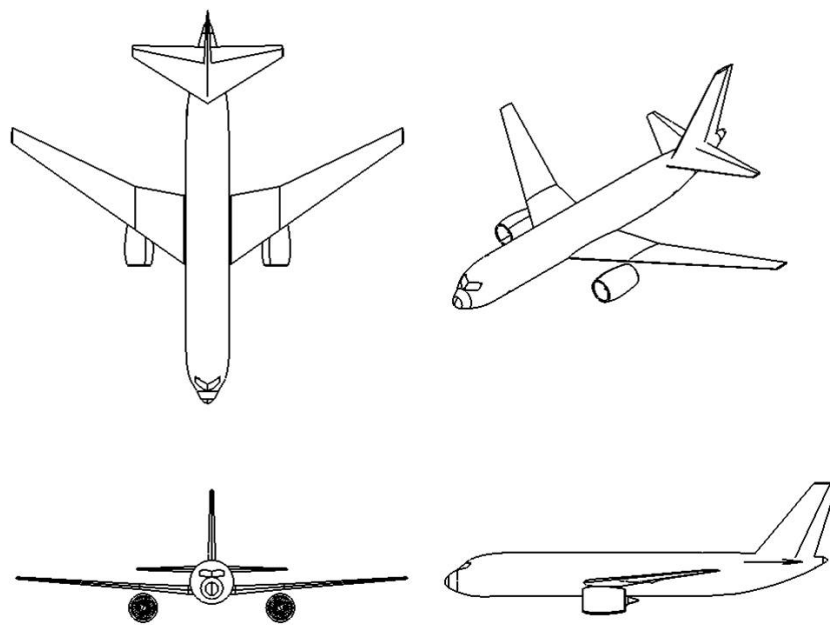


Figure B.2: Baseline Conventional Aircraft

Table B.1: Baseline Box Wing Aircraft Specifications

Item	Unit	Specification
External Dimensions		
Fore wing span	m	37.6
Fore wing aspect ratio		11.95
Fore wing gross area	m^2	118.32
Fore wing sweep angle	$^\circ$	40
Fore wing incidence (root)	$^\circ$	-1.6
Fore wing incidence (tip)	$^\circ$	-3
Elevon	m	2.7
Aft wing span	m	37.6
Aft wing gross area	m^2	118.32
Aft wing sweep angle	$^\circ$	-25
Aft wing incidence (center)	$^\circ$	-1.5
Aft wing incidence (root)	$^\circ$	-1.6
Aft wing incidence (tip)	$^\circ$	-0.6
Elevator	m	2.7
Overall fuselage length	m	46.00
Maximum fuselage diameter	m	5.60
Overall height	m	10.40
Wheel track	m	6.60
Wheelbase	m	23.40
Weights		
Operating empty mass	kg	68,200.00
Design payload	kg	29,700.00
Maximum payload	kg	32,000.00
Maximum takeoff weight	kg	126,670.00
Maximum landing weight	kg	98,000.00
Maximum fuel capacity	kg	44,000.00
Maximum passenger capacity		270
Performance		
Powerplant		CF6-80C2-B1F
Maximum take-off thrust	kN	2x254
Maximum cruise speed	Mach	0.8
Typical cruise altitude	ft	35,000.00
FAR landing field length	m	1648
FAR take-off field length	m	1204
Design mission range	nm	4000
Fuel/pax/nm		0.032
DOC/nm		24.03
DOC/nm/seat		0.089

Table B.2: Baseline Conventional Cantilever Wing Aircraft Specifications

Item	Unit	Specification
External Dimensions		
Wing span	m	47.00
Aspect ratio		9.33
Gross area	m^2	236.64
Wing sweep angle	$^\circ$	30
Wing incidence (root)	$^\circ$	-2.7
Wing incidence (tip)	$^\circ$	0
Overall fuselage length	m	46.00
Maximum fuselage diameter	m	5.60
Weights		
Operating empty mass	kg	69,900.00
Design payload	kg	29,700.00
Maximum payload	kg	32,000.00
Maximum takeoff weight	kg	136,000.00
Maximum landing weight	kg	121,000.00
Maximum fuel capacity	kg	75,500.00
Maximum passenger capacity		270
Performance		
Powerplant		CF6-80C2-B1F
Maximum take-off thrust	kN	2x254
Maximum cruise speed	Mach	0.8
Typical cruise altitude	ft	35,000.00
FAR landing field length	m	1561
FAR take-off field length	m	1216
Design mission range	nm	4000
Fuel/pax/nm		0.083
DOC/nm		27.48
DOC/nm/seat		0.102

AVL Models

C.1 Box Wing Aircraft AVL Text File

```

Mach 0.8000
IYsym IZsym Zsym
0 0 0.0000
Sref Cref Bref
118.4260 3.1496 37.6000
Xref Yref Zref
40.9950 0.0000 10.6497
CDp
0.0000
Surfaces
=====Aft
Yduplicate 4 0.00000 Aft
SURFACE
RAft
Nchord Cspace Nspan Sspace
12 2.0000 32 0.0000
SCALE
sX sY sZ
1.0000 1.0000 1.0000
TRANSLATE
dX dY dZ
0.0000 0.0000 0.0000
ANGLE
Ainc
0.0000
INDEX
Lsurf
1
=====Aft section 1
SECTION

```

```
Xle Yle Zle Chord Angle
30.3900 18.8000 5.7200 1.6000 -1.5000
AFILE
Airfoil definition
SC(2)-0012.dat
CLAF
CLaf = CLalpha / (2 * pi)
1.0000
=====Aft section 2
SECTION
Xle Yle Zle Chord Angle
32.4400 18.8000 9.8200 1.6000 -1.6000
AFILE
Airfoil definition
SC(2)-0412.dat
CLAF
CLaf = CLalpha / (2 * pi)
1.0000
CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
Basename aileron
Railer on 1.0000 0.7000 0.0000 0.0000 0.0000 -1
=====Aft section 3
SECTION
Xle Yle Zle Chord Angle
39.9200 2.7800 10.9400 4.3000 -2.0000
AFILE
Airfoil definition
SC(2)-0414.dat
CLAF
CLaf = CLalpha / (2 * pi)
1.0000
=====Aft section 4
SECTION
Xle Yle Zle Chord Angle
39.9200 0.0000 10.9400 4.3000 -2.0000
AFILE
Airfoil definition
SC(2)-0414.dat
CLAF
CLaf = CLalpha / (2 * pi)
1.0000 =====Aft (mirror)
SURFACE
L Aft
Nchord Cspace Nspan Sspace
```

```
12 2.0000 32 0.0000
SCALE
sX sY sZ
1.0000 1.0000 1.0000
TRANSLATE
dX dY dZ
0.0000 0.0000 0.0000
ANGLE
Ainc
0.0000
INDEX
Lsurf
1
=====Aft section 5
SECTION
Xle Yle Zle Chord Angle
39.9200 0.0000 10.9400 4.3000 -2.0000
AFILE
Airfoil definition
SC(2)-0414.dat
CLAF
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1.0000
=====Aft section 6
SECTION
Xle Yle Zle Chord Angle
39.9200 -2.7800 10.9400 4.3000 -2.0000
AFILE
Airfoil definition
SC(2)-0414.dat
CLAF
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1.0000
=====Aft section 7
SECTION
Xle Yle Zle Chord Angle
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AFILE
Airfoil definition
SC(2)-0412.dat
CLAF
 $CLaf = CLalpha / (2 * pi)$ 
1.0000
CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
```

Basename aileron

Laileron 1.0000 0.7000 0.0000 0.0000 0.0000 -1

=====Aft section 8

SECTION

Xle Yle Zle Chord Angle

30.3900 -18.8000 5.7200 1.6000 -1.5000

AFILE

Airfoil definition

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CLAF

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Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

0.0000 0.0000 0.8000

ANGLE

Ainc

0.0000

=====Vert section 1 SECTION

Xle Yle Zle Chord Angle

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AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

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1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====Vert section 2 SECTION

Xle Yle Zle Chord Angle

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AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

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1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====Vert (mirror) Ignore

SURFACE

LVert

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

0.0000 0.0000 0.8000

ANGLE

Ainc

0.0000

=====Vert section 3 SECTION

Xle Yle Zle Chord Angle

37.1200 -1.5000 4.8400 5.4000 0.0000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

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1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====Vert section 4 SECTION

Xle Yle Zle Chord Angle

39.9200 -2.7800 10.2000 2.8000 0.0000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

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1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====Fore Yduplicate 5 0.00000 Fore

SURFACE

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Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

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TRANSLATE

dX dY dZ

0.0000 0.0000 0.0000

ANGLE

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INDEX

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2

=====Fore section 1 SECTION

Xle Yle Zle Chord Angle

30.3900 18.8000 5.7200 1.6000 -0.9000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

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=====Fore section 2 SECTION

Xle Yle Zle Chord Angle

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AFILE

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=====Fore section 3 SECTION

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AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

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=====Fore section 4 SECTION

Xle Yle Zle Chord Angle

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AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

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1.0000

=====Fore section 5 SECTION

Xle Yle Zle Chord Angle

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AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

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1.0000

=====Fore (mirror) Ignore

SURFACE

LFore

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

0.0000 0.0000 0.0000

ANGLE

Ainc

0.0000

INDEX

Lsurf

2

=====Fore section 6 SECTION

Xle Yle Zle Chord Angle

14.9000 0.0000 0.5000 1.7200 -0.5000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 7 SECTION

Xle Yle Zle Chord Angle

14.9000 -2.7800 0.5000 5.1000 -2.3000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 8 SECTION

Xle Yle Zle Chord Angle

18.0300 -6.5000 0.7600 3.5300 -1.6000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 9 SECTION

Xle Yle Zle Chord Angle

28.3400 -18.8000 1.6200 1.6000 -2.0000

AFILE

Airfoil definition

SC(2)-0412.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 10 SECTION

Xle Yle Zle Chord Angle

30.3900 -18.8000 5.7200 1.6000 -0.9000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

***** Bodies *****

=====Fuselage BODY

Fuselage

Nbody Bspace

15 2.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

0.0000 0.0000 0.0000

BFILE
Body file
Fuselage.dat

C.2 Conventional Wing Aircraft AVL Text File

Mach
0.8000
IYsym IZsym Zsym
0 0 0.0000
Sref Cref Bref
Auto-generate
212.6365 4.5242 47.0000

Xref Yref Zref
Auto-generate
20.8300 0.0000 0.7759
CDp
0.0000

Surfaces
=====Fore
Yduplicate 5 0.00000 Fore
SURFACE
RFore
Nchord Cspace Nspan Sspace
12 2.0000 32 0.0000

SCALE
sX sY sZ
1.0000 1.0000 1.0000

TRANSLATE
dX dY dZ
20.0000 0.0000 1.0000

ANGLE
Ainc
0.0000

INDEX

Lsurf

1

=====Fore section 1 SECTION

Xle Yle Zle Chord Angle

14.4800 23.5000 0.7200 1.9000 0.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Aileron

RAileron 1.0000 0.7000 0.0000 0.0000 0.0000 -1

=====Fore section 2 SECTION

Xle Yle Zle Chord Angle

10.2900 17.5000 0.5100 2.9300 -2.5000

AFILE

Airfoil definition

SC(2)-0412.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Aileron

RAileron 1.0000 0.7000 0.0000 0.0000 0.0000 -1

=====Fore section 3 SECTION

Xle Yle Zle Chord Angle

4.1300 8.7000 0.2100 5.1950 -3.1000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 4 SECTION

Xle Yle Zle Chord Angle

0.0000 2.8000 0.0000 8.3000 -2.7000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 5 SECTION

Xle Yle Zle Chord Angle

0.0000 0.0000 0.0000 3.3200 0.0000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore (mirror) Ignore

SURFACE

LFore

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

20.0000 0.0000 1.0000

ANGLE

Ainc

0.0000

INDEX

Lsurf

1

=====Fore section 6 SECTION

Xle Yle Zle Chord Angle

0.0000 0.0000 0.0000 3.3200 0.0000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

 $CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 7 SECTION

Xle Yle Zle Chord Angle

0.0000 -2.8000 0.0000 8.3000 -2.7000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

 $CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 8 SECTION

Xle Yle Zle Chord Angle

4.1300 -8.7000 0.2100 5.1950 -3.1000

AFILE

Airfoil definition

SC(2)-0414.dat

CLAF

 $CLaf = CLalpha / (2 * pi)$

1.0000

=====Fore section 9 SECTION

Xle Yle Zle Chord Angle

10.2900 -17.5000 0.5100 2.9300 -2.5000

AFILE

Airfoil definition

SC(2)-0412.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Aileron

LAileron 1.0000 0.7000 0.0000 0.0000 0.0000 -1

=====Fore section 10

SECTION

Xle Yle Zle Chord Angle

14.4800 -23.5000 0.7200 1.9000 0.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Aileron

LAileron 1.0000 0.7000 0.0000 0.0000 0.0000 -1

=====fin

Yduplicate 2 0.00000 fin

SURFACE

Rfin

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

15.5000 0.0000 0.0000

ANGLE

Ainc
0.0000

INDEX
Lsurf
2

=====fin section 1 SECTION

Xle Yle Zle Chord Angle
24.0000 0.0000 4.5000 7.5000 0.0000

AFILE
Airfoil definition
SC(2)-0012.dat

CLAF
 $CLaf = CLalpha / (2 * pi)$
1.0000

CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
Basename Rudder
LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
Ignore
RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====fin section 2 SECTION

Xle Yle Zle Chord Angle
28.6300 0.0000 13.3600 2.8700 0.0000

AFILE
Airfoil definition
SC(2)-0012.dat

CLAF
 $CLaf = CLalpha / (2 * pi)$
1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Ignore

RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====fin (mirror) Ignore

SURFACE

Lfin

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

15.5000 0.0000 0.0000

ANGLE

Ainc

0.0000

INDEX

Lsurf

2

=====fin section 3 SECTION

Xle Yle Zle Chord Angle

28.6300 0.0000 13.3600 2.8700 0.0000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

 $CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Ignore

RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====fin section 4 SECTION

Xle Yle Zle Chord Angle

24.0000 0.0000 4.5000 7.5000 0.0000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Rudder

LRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Ignore

RRudder 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====tailplane

Yduplicate 2 0.00000 tailplane

SURFACE

Rtailplane

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ
1.0000 1.0000 1.0000

TRANSLATE
dX dY dZ
15.5000 0.0000 0.0000

ANGLE
Ainc
0.0000

INDEX
Lsurf
3

=====tailplane section 1

SECTION
Xle Yle Zle Chord Angle
24.0000 0.0000 4.5000 6.8000 0.5000

AFILE
Airfoil definition
SC(2)-0012.dat

CLAF
 $CLaf = CLalpha / (2 * pi)$
1.0000

CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
Basename Elevator
LElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL
label gain Xhinge Xhvec Yhvec Zhvec SgnDup
Ignore
RElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====tailplane section 2

SECTION
Xle Yle Zle Chord Angle
29.6200 9.0000 4.5000 1.3600 0.5000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

$CLaf = CLalpha / (2 * \pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Elevator

RElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====tailplane (mirror)

Ignore

SURFACE

Ltailplane

Nchord Cspace Nspan Sspace

12 2.0000 32 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

15.5000 0.0000 0.0000

ANGLE

Ainc

0.0000

INDEX

Lsurf

3

=====tailplane section 3

SECTION

Xle Yle Zle Chord Angle

29.6200 -9.0000 4.5000 1.3600 0.5000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Elevator

LElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

=====tailplane section 4

SECTION

Xle Yle Zle Chord Angle

24.0000 0.0000 4.5000 6.8000 0.5000

AFILE

Airfoil definition

SC(2)-0012.dat

CLAF

$CLaf = CLalpha / (2 * pi)$

1.0000

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Basename Elevator

LElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

CONTROL

label gain Xhinge Xhvec Yhvec Zhvec SgnDup

Ignore

RElevator 1.0000 0.7000 0.0000 0.0000 0.0000 1

***** Bodies *****

=====Fuselage

BODY

Fuselage

Nbody Bspace

15 0.0000

SCALE

sX sY sZ

1.0000 1.0000 1.0000

TRANSLATE

dX dY dZ

0.0000 0.0000 0.0000

BFILE

Body file

fuselage.dat

Aircraft Mass Statements

D.1 Box Wing Aircraft Mass Statements

Table D.1: Box Wing - OEM

Component	Mass (kg)	x (m)	Mx (kgm)
Crew	611.25	7.00	4278.75
Operational Items	2970	7.00	20790.00
Fuselage	11566.48	21.62	250067.32
Fore Wing	5090.95	18.92	96299.07
Aft Wing	4947.02	36.08	178491.29
Tip Fin	763.64	26.32	20096.30
Nacelles	2362.40	36.50	86227.70
MLG	4323.37	28.60	123648.51
NLG	646.02	5.00	3230.11
FW Control surfaces	432.15	19.52	8433.76
AW Control surfaces	432.15	35.48	15332.98
Tail Structure	3563.67	34.30	122233.74
Propulsion Mass	7329.48	37.00	271190.78
Avionics	8796.40	6.13	53916.78
APU	3769.89	43.40	163613.08
Subtotal	57604.89		1417850.16
x cg		24.61	

The total payload for the study was computed as shown below:

Payload = total number of passengers x 75.00kg per passenger⁵⁴ + total number of passengers x average baggage per passenger = 31050.00kg

Table D.2: Box Wing - MTOM

<i>Component</i>	<i>Mass</i>	<i>x</i>	<i>Mx</i>
	(kg)	(m)	(kgm)
Crew	611.25	7.00	4278.75
Operational Items	2970	7.00	20790.00
Fuselage	11566.48	21.62	250067.32
Forward Payload Bay	6831.00	11.90	81288.90
Forward Fuel	13292.47	16.92	224851.87
Fore Wing	5090.95	18.92	96299.07
Aft Wing	4947.02	36.08	178491.29
Tip Fin	763.64	26.32	20096.30
Nacelles	2362.40	36.50	86227.70
MLG	4323.37	28.60	123648.51
NLG	646.02	5.00	3230.11
FW Control surfaces	432.15	19.52	8433.76
AW Control surfaces	432.15	35.48	15332.98
Tail Structure	3563.67	34.30	122233.74
Propulsion Mass	7329.48	37.00	271190.78
Avionics	8796.40	6.13	53916.78
APU	3769.89	43.40	163613.08
Aft Payload Bay	24219.00	26.80	649069.20
Aft Fuel	12291.63	34.58	425051.19
Subtotal	114238.99		2798111.32
x cg		24.49	

D.2 Conventional Cantilever Aircraft Mass Statements

Table D.3: Cantilever Wing - OEM

Component	Mass	x	Mx
	(kg)	(m)	(kgm)
Crew	611.25	9.00	5501.25
Operational Items	2970	13.50	40095.00
Fuselage	11143.32	19.32	215288.88
Wing	8028.22	23.97	192458.02
Nacelles	2379.18	16.20	38542.65
MLG	4354.07	25.50	111028.80
NLG	650.61	5.00	3253.04
Wing Control surfaces	1157.99	24.57	28455.04
Tail Structure	2027.10	42.05	85239.66
Propulsion Mass	7381.52	16.70	123271.39
Avionics	8858.86	27.19	240849.91
APU	3796.65	45.00	170849.38
Subtotal	53358.77		1254833.02
x cg		23.52	

Table D.4: Cantilever Wing - MTOM

Component	Mass	x	Mx
	(kg)	(m)	(kgm)
Crew	611.25	9.00	5501.25
Forward Payload Bay	12730.5	13.00	165496.50
Operational Items	2970	13.50	40095.00
Fuselage	11143.32	19.32	215288.88
Wing	8028.22	23.97	192458.02
Nacelles	2379.18	16.20	38542.65
MLG	4354.07	25.50	111028.80
NLG	650.61	5.00	3253.04
Wing Control surfaces	1157.99	24.57	28455.04
Tail Structure	2027.10	42.05	85239.66
Propulsion Mass	7381.52	16.70	123271.39
Avionics	8858.86	27.19	240849.91
APU	3796.65	45.00	170849.38
Aft Payload Bay	18319.50	30.00	549585.00
Fuel	30641.32	22.70	695557.87
Subtotal	115050.09		2665472.39
x cg		23.17	

Aircraft Component Inertias

Table E.1: Box Wing Aircraft Mass and Inertia Samples

<i>Parameter</i>	<i>Case One</i>	<i>Case Two</i>	<i>Case Three</i>	<i>Case Four</i>
<i>Mass</i> (kg)	57604.89	87431.99	100635.44	114238.99
<i>Fuel</i> (kg)	0	19581	22538	25584
<i>Payload</i> (kg)	0	10247	20493	31050
<i>xbar</i> (m)	24.61	24.57	24.57	24.49
<i>ybar</i> (m)	0.00	0.00	0.00	0.00
<i>zbar</i> (m)	1.02	1.07	0.92	0.79
<i>Ixx</i> (kgm ²)	1410388	3617727	4311877	4302164
<i>Iyy</i> (kgm ²)	9176974	12675955	13874526	15196470
<i>Izz</i> (kgm ²)	9504931	13905435	15610760	16672538
<i>h</i>	0.13	0.12	0.12	0.11

Where:

Case One	= <i>OEM</i>	
Case Two	= <i>OEM</i>	plus 33% payload
Case Three	= <i>OEM</i>	plus 66% payload
Case Four	= <i>M_{TOM}</i>	

Table E.2: Conventional Cantilever Aircraft Mass and Inertia Samples

<i>Parameter</i>	<i>Case One</i>	<i>Case Two</i>	<i>Case Three</i>	<i>Case Four</i>
<i>Mass</i> (kg)	53358.77	89035.46	103271.16	115050.09
<i>Fuel</i> (kg)	0	25430	29419	30641
<i>Payload</i> (kg)	0	10247	20493	31050
<i>xbar</i> (m)	23.52	23.44	23.29	23.17
<i>ybar</i> (m)	0.00	0.00	0.00	0.00
<i>zbar</i> (m)	-0.68	-0.50	-0.51	-0.53
<i>Ixx</i> (kgm ²)	1484252	1794270	1861052	4506463
<i>Iyy</i> (kgm ²)	4282283	5586658	6953905	10039598
<i>Izz</i> (kgm ²)	5388866	6953337	8347346	14046751
<i>h</i>	0.49	0.47	0.44	0.41

Where:

Case One = *OEM*Case Two = *OEM* plus 33% payloadCase Three = *OEM* plus 66% payloadCase Four = *M_{TOM}*

APPENDIX **F**

Aircraft Aerodynamic Data

Angle of Attack	-16	-14	-12	-10	-8	-6	-4	-2	0	2	4	6	8	10	12	14	16
Elevon Deflection																	
-16	-1.028	-1.030	-1.034	-1.018	-0.970	-0.942	-0.824	-0.748	-0.594	-0.429	-0.229	-0.117	0.023	0.159	0.292	0.427	0.531
-14	-0.997	-1.002	-0.998	-0.998	-0.914	-0.873	-0.776	-0.629	-0.490	-0.319	-0.146	-0.020	0.123	0.267	0.413	0.528	0.622
-12	-0.970	-0.967	-0.946	-0.911	-0.850	-0.766	-0.662	-0.543	-0.382	-0.174	-0.061	0.084	0.238	0.396	0.522	0.626	0.709
-10	-0.957	-0.940	-0.898	-0.852	-0.777	-0.685	-0.569	-0.434	-0.274	-0.078	0.047	0.206	0.370	0.512	0.629	0.721	0.790
-8	-0.927	-0.891	-0.881	-0.781	-0.700	-0.624	-0.469	-0.317	-0.135	0.024	0.167	0.335	0.496	0.624	0.729	0.807	0.868
-6	-0.884	-0.831	-0.782	-0.713	-0.622	-0.522	-0.350	-0.183	0.018	0.129	0.301	0.472	0.613	0.730	0.819	0.887	0.933
-4	-0.814	-0.762	-0.707	-0.615	-0.506	-0.378	-0.222	-0.048	0.143	0.258	0.442	0.595	0.722	0.825	0.901	0.960	0.996
-2	-0.753	-0.730	-0.611	-0.516	-0.425	-0.275	-0.092	0.115	0.218	0.402	0.568	0.707	0.836	0.926	0.992	1.039	1.062
0	-0.681	-0.609	-0.520	-0.425	-0.302	-0.133	0.056	0.229	0.314	0.538	0.689	0.816	0.920	0.995	1.050	1.082	1.094
2	-0.592	-0.537	-0.452	-0.302	-0.160	0.009	0.203	0.297	0.503	0.648	0.787	0.901	0.989	1.057	1.098	1.121	1.124
4	-0.512	-0.424	-0.342	-0.202	-0.033	0.134	0.288	0.349	0.614	0.748	0.877	0.976	1.053	1.105	1.136	1.149	1.143
6	-0.426	-0.327	-0.213	-0.073	0.080	0.231	0.348	0.549	0.716	0.839	0.951	1.039	1.106	1.148	1.171	1.174	1.160
8	-0.334	-0.228	-0.104	0.032	0.181	0.335	0.404	0.655	0.769	0.921	1.021	1.098	1.152	1.183	1.196	1.190	1.170
10	-0.247	-0.136	-0.009	0.130	0.267	0.391	0.580	0.716	0.895	0.993	1.082	1.151	1.193	1.215	1.219	1.201	1.174
12	-0.158	-0.046	0.081	0.209	0.344	0.448	0.639	0.802	0.923	1.062	1.139	1.194	1.227	1.239	1.234	1.212	1.174
14	-0.076	0.038	0.160	0.304	0.452	0.508	0.742	0.882	0.996	1.120	1.188	1.232	1.253	1.258	1.243	1.215	1.171
16	0.004	0.117	0.255	0.386	0.484	0.675	0.825	0.954	1.041	1.177	1.233	1.266	1.276	1.272	1.249	1.215	1.167

Figure F.1: Fore Wing Lift Coefficient Variation

Angle of Attack	-16	-14	-12	-10	-8	-6	-4	-2	0	2	4	6	8	10	12	14	16
Elevator deflection																	
-16	-0.156	-0.156	-0.157	-0.155	-0.147	-0.143	-0.125	-0.114	-0.090	-0.065	-0.035	-0.018	0.003	0.024	0.044	0.065	0.081
-14	-0.151	-0.152	-0.151	-0.151	-0.139	-0.133	-0.118	-0.095	-0.074	-0.048	-0.022	-0.003	0.019	0.041	0.063	0.080	0.094
-12	-0.147	-0.147	-0.144	-0.138	-0.129	-0.116	-0.100	-0.082	-0.058	-0.026	-0.009	0.013	0.036	0.060	0.079	0.095	0.108
-10	-0.145	-0.143	-0.136	-0.129	-0.118	-0.104	-0.086	-0.066	-0.042	-0.012	0.007	0.031	0.056	0.078	0.095	0.109	0.120
-8	-0.141	-0.135	-0.134	-0.119	-0.106	-0.095	-0.071	-0.048	-0.020	0.004	0.025	0.051	0.075	0.095	0.111	0.123	0.132
-6	-0.134	-0.126	-0.119	-0.108	-0.094	-0.079	-0.053	-0.028	0.003	0.020	0.046	0.072	0.093	0.111	0.124	0.135	0.142
-4	-0.124	-0.116	-0.107	-0.093	-0.077	-0.057	-0.034	-0.007	0.022	0.039	0.067	0.090	0.110	0.125	0.137	0.146	0.151
-2	-0.114	-0.111	-0.093	-0.078	-0.065	-0.042	-0.014	0.017	0.033	0.061	0.086	0.107	0.127	0.141	0.151	0.158	0.161
0	-0.103	-0.092	-0.079	-0.065	-0.046	-0.020	0.009	0.035	0.048	0.082	0.105	0.124	0.140	0.151	0.159	0.164	0.166
2	-0.090	-0.082	-0.069	-0.046	-0.024	0.001	0.031	0.045	0.076	0.098	0.119	0.137	0.150	0.160	0.167	0.170	0.171
4	-0.078	-0.064	-0.052	-0.031	-0.005	0.020	0.044	0.053	0.093	0.114	0.133	0.148	0.160	0.168	0.172	0.174	0.174
6	-0.065	-0.050	-0.032	-0.011	0.012	0.035	0.053	0.083	0.109	0.127	0.144	0.158	0.168	0.174	0.178	0.178	0.176
8	-0.051	-0.035	-0.016	0.005	0.027	0.051	0.061	0.099	0.117	0.140	0.155	0.167	0.175	0.180	0.182	0.181	0.178
10	-0.037	-0.021	-0.001	0.020	0.041	0.059	0.088	0.109	0.136	0.151	0.164	0.175	0.181	0.184	0.185	0.182	0.178
12	-0.024	-0.007	0.012	0.032	0.052	0.068	0.097	0.122	0.140	0.161	0.173	0.181	0.186	0.188	0.187	0.184	0.178
14	-0.012	0.006	0.024	0.046	0.069	0.077	0.113	0.134	0.151	0.170	0.180	0.187	0.190	0.191	0.189	0.184	0.178
16	0.001	0.018	0.039	0.059	0.073	0.102	0.125	0.145	0.158	0.179	0.187	0.192	0.194	0.193	0.190	0.184	0.177

Figure F.2: Aft Wing Lift Coefficient Variation

AoA	-16	-14	-12	-10	-8	-6	-4	-2	0	2	4	6	8	10	12	14	16
Elevon deflection																	
-16	0.020	0.020	0.020	0.019	0.017	0.016	0.013	0.010	0.007	0.003	0.001	0.000	0.000	0.000	0.002	0.003	0.005
-14	0.018	0.019	0.018	0.018	0.015	0.014	0.011	0.007	0.004	0.002	0.000	0.000	0.000	0.001	0.003	0.005	0.007
-12	0.017	0.017	0.017	0.015	0.013	0.011	0.008	0.005	0.003	0.001	0.000	0.000	0.001	0.003	0.005	0.007	0.009
-10	0.017	0.016	0.015	0.013	0.011	0.009	0.006	0.003	0.001	0.000	0.000	0.001	0.003	0.005	0.007	0.010	0.012
-8	0.016	0.015	0.014	0.011	0.009	0.007	0.004	0.002	0.000	0.000	0.001	0.002	0.005	0.007	0.010	0.012	0.014
-6	0.014	0.013	0.011	0.009	0.007	0.005	0.002	0.001	0.000	0.000	0.002	0.004	0.007	0.010	0.012	0.015	0.016
-4	0.012	0.011	0.009	0.007	0.005	0.003	0.001	0.000	0.000	0.001	0.004	0.007	0.010	0.013	0.015	0.017	0.018
-2	0.010	0.010	0.007	0.005	0.003	0.001	0.000	0.000	0.001	0.003	0.006	0.009	0.013	0.016	0.018	0.020	0.021
0	0.009	0.007	0.005	0.003	0.002	0.000	0.000	0.001	0.002	0.005	0.009	0.012	0.016	0.018	0.020	0.022	0.022
2	0.006	0.005	0.004	0.002	0.000	0.000	0.001	0.002	0.005	0.008	0.011	0.015	0.018	0.021	0.022	0.023	0.023
4	0.005	0.003	0.002	0.001	0.000	0.000	0.002	0.002	0.007	0.010	0.014	0.018	0.020	0.023	0.024	0.024	0.024
6	0.003	0.002	0.001	0.000	0.000	0.001	0.002	0.006	0.009	0.013	0.017	0.020	0.023	0.024	0.025	0.025	0.025
8	0.002	0.001	0.000	0.000	0.001	0.002	0.003	0.008	0.011	0.016	0.019	0.022	0.025	0.026	0.026	0.026	0.025
10	0.001	0.000	0.000	0.000	0.001	0.003	0.006	0.009	0.015	0.018	0.022	0.024	0.026	0.027	0.027	0.027	0.025
12	0.000	0.000	0.000	0.001	0.002	0.004	0.008	0.012	0.016	0.021	0.024	0.026	0.028	0.028	0.028	0.027	0.025
14	0.000	0.000	0.000	0.002	0.004	0.005	0.010	0.014	0.018	0.023	0.026	0.028	0.029	0.029	0.029	0.027	0.025
16	0.000	0.000	0.001	0.003	0.004	0.008	0.013	0.017	0.020	0.026	0.028	0.030	0.030	0.030	0.029	0.027	0.025

Figure F.3: Fore Wing Trim Drag

AoA	-16	-14	-12	-10	-8	-6	-4	-2	0	2	4	6	8	10	12	14	16
Elevator deflection																	
-16	0.019	0.019	0.019	0.018	0.017	0.016	0.012	0.010	0.006	0.003	0.001	0.000	0.000	0.000	0.001	0.003	0.005
-14	0.017	0.018	0.017	0.017	0.015	0.013	0.011	0.007	0.004	0.002	0.000	0.000	0.000	0.001	0.003	0.005	0.007
-12	0.017	0.016	0.016	0.015	0.013	0.010	0.008	0.005	0.003	0.001	0.000	0.000	0.001	0.003	0.005	0.007	0.009
-10	0.016	0.016	0.014	0.013	0.011	0.008	0.006	0.003	0.001	0.000	0.000	0.000	0.001	0.002	0.005	0.009	0.011
-8	0.015	0.014	0.014	0.011	0.009	0.007	0.004	0.002	0.000	0.000	0.000	0.000	0.002	0.004	0.007	0.011	0.013
-6	0.014	0.012	0.011	0.009	0.007	0.005	0.002	0.001	0.000	0.000	0.002	0.004	0.007	0.009	0.012	0.014	0.015
-4	0.012	0.010	0.009	0.007	0.004	0.003	0.001	0.000	0.000	0.001	0.003	0.006	0.009	0.012	0.014	0.016	0.017
-2	0.010	0.009	0.007	0.005	0.003	0.001	0.000	0.000	0.001	0.003	0.006	0.009	0.012	0.015	0.017	0.019	0.020
0	0.008	0.007	0.005	0.003	0.002	0.000	0.000	0.001	0.002	0.005	0.008	0.012	0.015	0.017	0.019	0.021	0.021
2	0.006	0.005	0.004	0.002	0.000	0.000	0.001	0.002	0.004	0.007	0.011	0.014	0.017	0.020	0.021	0.022	0.022
4	0.005	0.003	0.002	0.001	0.000	0.000	0.001	0.002	0.007	0.010	0.013	0.017	0.019	0.021	0.023	0.023	0.023
6	0.003	0.002	0.001	0.000	0.000	0.001	0.002	0.005	0.009	0.012	0.016	0.019	0.021	0.023	0.024	0.024	0.024
8	0.002	0.001	0.000	0.000	0.001	0.002	0.003	0.008	0.010	0.015	0.018	0.021	0.023	0.025	0.025	0.025	0.024
10	0.001	0.000	0.000	0.000	0.001	0.003	0.006	0.009	0.014	0.017	0.021	0.023	0.025	0.026	0.026	0.025	0.024
12	0.000	0.000	0.000	0.001	0.002	0.004	0.007	0.011	0.015	0.020	0.023	0.025	0.026	0.027	0.027	0.026	0.024
14	0.000	0.000	0.000	0.002	0.004	0.005	0.010	0.014	0.017	0.022	0.025	0.027	0.028	0.028	0.027	0.026	0.024
16	0.000	0.000	0.001	0.003	0.004	0.008	0.012	0.016	0.019	0.024	0.027	0.028	0.029	0.028	0.027	0.026	0.024

Figure F.4: Aft Wing Trim Drag

AoA	-16	-14	-12	-10	-8	-6	-4	-2	0	2	4	6	8	10	12	14	16
Elevator deflection																	
-16	0.19	0.173	0.158	0.144	0.127	0.117	0.099	0.092	0.08	0.077	0.056	0.056	0.056	0.048	0.035	0.013	-0.009
-14	0.171	0.155	0.14	0.128	0.108	0.097	0.083	0.069	0.06	0.055	0.04	0.039	0.034	0.021	0.001	-0.019	-0.042
-12	0.154	0.136	0.119	0.103	0.087	0.072	0.059	0.049	0.04	0.024	0.022	0.019	0.007	-0.012	-0.032	-0.052	-0.073
-10	0.137	0.118	0.099	0.081	0.065	0.05	0.037	0.026	0.018	0.009	-0.001	-0.009	-0.027	-0.045	-0.067	-0.085	-0.111
-8	0.116	0.095	0.08	0.058	0.041	0.028	0.013	0.001	-0.008	-0.021	-0.027	-0.038	-0.06	-0.081	-0.099	-0.119	-0.144
-6	0.093	0.071	0.052	0.034	0.017	0	-0.015	-0.029	-0.03	-0.036	-0.063	-0.08	-0.097	-0.116	-0.135	-0.149	-0.175
-4	0.065	0.044	0.024	0.004	-0.015	-0.033	-0.048	-0.065	-0.075	-0.084	-0.095	-0.108	-0.13	-0.147	-0.166	-0.187	-0.206
-2	0.038	0.016	-0.007	-0.028	-0.052	-0.071	-0.082	-0.072	-0.097	-0.115	-0.131	-0.139	-0.165	-0.175	-0.199	-0.218	-0.235
0	0.006	-0.018	-0.04	-0.064	-0.089	-0.103	-0.108	-0.116	-0.11	-0.151	-0.157	-0.175	-0.19	-0.205	-0.221	-0.229	-0.255
2	-0.027	-0.054	-0.081	-0.096	-0.109	-0.123	-0.124	-0.141	-0.167	-0.174	-0.173	-0.197	-0.206	-0.222	-0.228	-0.248	-0.274
4	-0.061	-0.08	-0.112	-0.135	-0.15	-0.157	-0.155	-0.124	-0.188	-0.182	-0.194	-0.216	-0.214	-0.243	-0.251	-0.267	-0.289
6	-0.089	-0.112	-0.137	-0.161	-0.167	-0.141	-0.166	-0.202	-0.206	-0.201	-0.21	-0.233	-0.246	-0.245	-0.274	-0.27	-0.304
8	-0.121	-0.142	-0.162	-0.179	-0.193	-0.145	-0.163	-0.219	-0.209	-0.229	-0.238	-0.234	-0.244	-0.259	-0.272	-0.283	-0.3
10	-0.154	-0.18	-0.17	-0.208	-0.165	-0.112	-0.232	-0.219	-0.237	-0.227	-0.252	-0.249	-0.259	-0.272	-0.28	-0.297	-0.308
12	-0.177	-0.185	-0.216	-0.207	-0.204	-0.184	-0.225	-0.231	-0.235	-0.242	-0.25	-0.26	-0.271	-0.283	-0.295	-0.307	-0.317
14	-0.208	-0.209	-0.188	-0.232	-0.237	-0.173	-0.241	-0.243	-0.248	-0.252	-0.256	-0.27	-0.282	-0.292	-0.304	-0.314	-0.324
16	-0.228	-0.223	-0.247	-0.238	-0.201	-0.252	-0.251	-0.253	-0.252	-0.264	-0.271	-0.282	-0.288	-0.295	-0.311	-0.32	-0.329

Figure F.5: Pitching Moment

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	503913	473861	447003	423160	402152	383799	367922	354342	342879	333353	325587	319399	314466
0.9529	2000	489535	460341	434249	411086	390677	372848	357424	344231	333096	323842	316297	310286	305494
0.9075	4000	475405	447053	421715	399220	379400	362086	347107	334295	323481	314494	307167	301329	296676
0.8217	6000	447902	421191	397319	376125	357452	341139	327027	314956	304767	296301	289397	283897	279513
0.7812	8000	434523	408610	385450	364890	346774	330949	317258	305548	295664	287450	280753	275417	271164
0.7423	10000	421408	396277	373817	353877	336308	320960	307683	296326	286740	278774	272279	267104	262979
0.7048	12000	408503	384141	362368	343040	326009	311131	298260	287251	277959	270237	263941	258924	254926
0.6689	14000	395888	372278	351178	332446	315941	301523	289050	278381	269375	261892	255790	250929	247053
0.6012	16000	371336	349191	329399	311829	296347	282823	271123	261116	252669	245650	239926	235367	231732
0.5694	18000	359423	337988	318832	301825	286840	273750	262425	252739	244563	237769	232229	227816	224297
0.5389	20000	347744	327006	308472	292018	277520	264855	253899	244527	236617	230043	224684	220414	217010
0.5096	22000	336274	316220	298297	282385	268366	256119	245524	236461	228812	222455	217272	213143	209851
0.4549	24000	314127	295393	278651	263788	250691	239251	229353	220888	213742	207804	202963	199105	196030
0.4292	26000	303355	285264	269096	254742	242095	231046	221489	213313	206413	200678	196003	192278	189308
0.4047	28000	292843	275379	259771	245915	233706	223040	213814	205922	199260	193725	189211	185615	182749
0.3813	30000	282563	265712	250652	237282	225502	215211	206308	198693	192265	186924	182569	179099	176333
0.3589	31000	272483	256233	241710	228817	217457	207533	198948	191605	185406	180256	176056	172710	170043

Figure F.6: Engine Maximum Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	438185	412053	388699	367965	349697	333738	319932	308123	298155	289873	283119	277738	273449
0.9529	2000	425683	400296	377608	357466	339719	324216	310804	299332	289648	281602	275041	269814	265647
0.9075	4000	413395	388742	366709	347148	329913	314857	301832	290691	281288	273473	267102	262025	257979
0.8217	6000	389480	366253	345494	327066	310828	296643	284371	273875	265015	257653	251650	246867	243055
0.7812	8000	377846	355313	335174	317296	301543	287782	275877	265694	257099	249957	244133	239493	235794
0.7423	10000	366442	344589	325058	307719	292442	279096	267550	257675	249339	242412	236764	232265	228678
0.7048	12000	355220	334036	315103	298295	283486	270549	259357	249784	241703	234989	229514	225152	221674
0.6689	14000	344250	323720	305372	289084	274732	262194	251347	242070	234239	227732	222426	218199	214829
0.6012	16000	322901	303644	286434	271155	257693	245933	235759	227057	219712	213609	208632	204667	201506
0.5694	18000	312542	293903	277245	262456	249426	238043	228196	219773	212664	206756	201939	198101	195041
0.5389	20000	302387	284353	268237	253929	241322	230309	220781	212632	205754	200038	195377	191664	188704
0.5096	22000	292412	274974	259388	245553	233362	222712	213499	205618	198967	193439	188932	185342	182479
0.4549	24000	273154	256864	242305	229380	217992	208044	199438	192076	185863	180699	176489	173135	170461
0.4292	26000	263787	248056	233996	221515	210517	200910	192599	185490	179489	174503	170437	167198	164616
0.4047	28000	254646	239460	225888	213839	203223	193948	185925	179062	173270	168456	164531	161404	158912
0.3813	30000	245707	231054	217958	206332	196088	187140	179398	172776	167187	162543	158756	155738	153333
0.3589	31000	236942	222811	210183	198972	189093	180464	172998	166613	161223	156744	153092	150183	147863

Figure F.7: Engine Takeoff Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	100157	94184	88845	84106	79931	76283	73127	70428	68150	66257	64713	63483	62503
0.9529	2000	97299	91496	86310	81707	77650	74106	71041	68419	66205	64366	62866	61672	60719
0.9075	4000	94490	88855	83819	79348	75409	71967	68990	66444	64294	62508	61052	59892	58967
0.8217	6000	89024	83715	78970	74758	71046	67804	64999	62600	60575	58892	57520	56427	55555
0.7812	8000	86365	81214	76611	72525	68924	65779	63058	60730	58765	57133	55802	54741	53896
0.7423	10000	83758	78763	74299	70336	66844	63793	61154	58897	56992	55409	54118	53089	52269
0.7048	12000	81193	76351	72024	68182	64797	61840	59281	57093	55246	53712	52460	51463	50668
0.6689	14000	78686	73993	69799	66076	62796	59930	57451	55330	53540	52053	50840	49874	49104
0.6012	16000	73806	69404	65471	61978	58901	56213	53888	51899	50220	48825	47687	46781	46058
0.5694	18000	71438	67178	63370	59990	57012	54410	52159	50234	48609	47258	46157	45280	44581
0.5389	20000	69117	64995	61311	58041	55159	52642	50464	48602	47029	45723	44658	43809	43132
0.5096	22000	66837	62851	59289	56126	53340	50906	48800	46998	45478	44215	43185	42364	41710
0.4549	24000	62435	58712	55384	52430	49827	47553	45586	43903	42483	41303	40340	39574	38963
0.4292	26000	60294	56698	53485	50632	48118	45922	44023	42398	41026	39886	38957	38217	37627
0.4047	28000	58205	54734	51632	48877	46451	44331	42497	40929	39604	38504	37607	36892	36323
0.3813	30000	56162	52812	49819	47162	44820	42775	41005	39492	38214	37153	36287	35597	35048
0.3589	31000	54158	50928	48042	45479	43221	41249	39542	38083	36851	35827	34992	34327	33797

Figure F.8: Engine Cruise Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	80125	75347	71076	67285	63945	61026	58502	56343	54520	53005	51770	50786	50002
0.9529	2000	77839	73197	69048	65365	62120	59285	56833	54735	52964	51493	50293	49337	48575
0.9075	4000	75592	71084	67055	63479	60327	57574	55192	53155	51435	50007	48841	47913	47173
0.8217	6000	71219	66972	63176	59806	56837	54243	51999	50080	48460	47114	46016	45141	44444
0.7812	8000	69092	64971	61289	58020	55139	52623	50446	48584	47012	45706	44641	43793	43117
0.7423	10000	67007	63010	59439	56269	53475	51035	48923	47118	45593	44327	43294	42471	41815
0.7048	12000	64954	61081	57619	54545	51837	49472	47425	45675	44197	42969	41968	41171	40535
0.6689	14000	62949	59195	55840	52861	50237	47944	45961	44264	42832	41642	40672	39899	39283
0.6012	16000	59045	55523	52376	49583	47121	44971	43110	41519	40176	39060	38150	37425	36847
0.5694	18000	57150	53742	50696	47992	45609	43528	41727	40187	38887	37807	36926	36224	35665
0.5389	20000	55294	51996	49049	46433	44127	42114	40371	38881	37624	36578	35726	35047	34506
0.5096	22000	53470	50281	47431	44901	42672	40724	39040	37599	36382	35372	34548	33891	33368
0.4549	24000	49948	46969	44307	41944	39861	38042	36469	35123	33986	33042	32272	31659	31170
0.4292	26000	48235	45359	42788	40506	38495	36738	35218	33918	32821	31909	31166	30573	30101
0.4047	28000	46564	43787	41305	39102	37161	35465	33998	32743	31684	30803	30086	29514	29058
0.3813	30000	44929	42250	39855	37729	35856	34220	32804	31593	30571	29722	29030	28478	28038
0.3589	31000	43326	40743	38433	36383	34577	32999	31634	30466	29481	28662	27994	27462	27038

Figure F.9: Engine Cruise Thrust (80%) as function of Altitude and Mach Number

Wing AoA	-5	-4	-3	-2	-1	0	1	2	3	4	5	6	7	8	9	10
Wing CL	-0.575	-0.460	-0.345	-0.230	-0.115	0.000	0.115	0.230	0.345	0.460	0.575	0.690	0.804	0.919	1.034	1.149

Figure F.10: Cantilever Wing Lift Coefficient Variation

Tailplane AoA	-12	-10	-8	-6	-4	-2	0	2	4	6
Elevator deflection										
-16	-1.135	-1.177	-1.157	-1.147	-1.086	-1.04	-0.94	-0.77	-0.62	-0.4
-14	-1.148	-1.129	-1.143	-1.091	-1.053	-0.95	-0.86	-0.68	-0.52	-0.32
-12	-1.129	-1.12	-1.093	-1.033	-0.99	-0.896	-0.77	-0.59	-0.36	-0.22
-10	-1.082	-1.091	-1.058	-1.002	-0.921	-0.813	-0.67	-0.48	-0.28	-0.11
-8	-1.044	-1.05	-0.992	-0.909	-0.842	-0.696	-0.55	-0.37	-0.17	0.01
-6	-1.004	-0.99	-0.938	-0.858	-0.728	-0.618	-0.44	-0.21	-0.05	0.147
-4	-0.956	-0.937	-0.875	-0.779	-0.657	-0.486	-0.32	-0.11	0.109	0.284
-2	-0.936	-0.853	-0.794	-0.655	-0.548	-0.356	-0.17	0.058	0.253	0.414
0	-0.85	-0.789	-0.66	-0.536	-0.388	-0.229	0	0.212	0.388	0.535
2	-0.784	-0.658	-0.585	-0.444	-0.27	-0.058	0.178	0.355	0.517	0.652
4	-0.656	-0.554	-0.463	-0.307	-0.109	0.109	0.329	0.486	0.631	0.75
6	-0.555	-0.447	-0.338	-0.153	0.051	0.214	0.455	0.595	0.727	0.833
8	-0.489	-0.359	-0.181	-0.01	0.168	0.37	0.57	0.696	0.815	0.909
10	-0.346	-0.212	-0.058	0.106	0.275	0.482	0.676	0.789	0.896	0.975
12	-0.241	-0.098	0.052	0.237	0.358	0.621	0.733	0.874	0.966	1.033
14	-0.147	0.006	0.169	0.315	0.529	0.68	0.812	0.95	1.029	1.084
16	-0.038	0.106	0.257	0.461	0.629	0.775	0.891	1.015	1.086	1.129

Figure F.11: Cantilever Aircraft Tailplane Lift Coefficient Variation

Tailplane AoA	-12	-10	-8	-6	-4	-2	0	2	4	6
Elevator deflection										
-16	-0.008	-0.009	-0.008	-0.008	-0.008	-0.008	-0.007	-0.006	-0.005	-0.003
-14	-0.008	-0.008	-0.008	-0.008	-0.008	-0.007	-0.006	-0.005	-0.004	-0.002
-12	-0.008	-0.008	-0.008	-0.008	-0.007	-0.007	-0.006	-0.004	-0.003	-0.002
-10	-0.008	-0.008	-0.008	-0.007	-0.007	-0.006	-0.005	-0.004	-0.002	-0.001
-8	-0.008	-0.008	-0.007	-0.007	-0.006	-0.005	-0.004	-0.003	-0.001	0.000
-6	-0.007	-0.007	-0.007	-0.006	-0.005	-0.005	-0.003	-0.002	0.000	0.001
-4	-0.007	-0.007	-0.006	-0.006	-0.005	-0.004	-0.002	-0.001	0.001	0.002
-2	-0.007	-0.006	-0.006	-0.005	-0.004	-0.003	-0.001	0.000	0.002	0.003
0	-0.006	-0.006	-0.005	-0.004	-0.003	-0.002	0.000	0.002	0.003	0.004
2	-0.006	-0.005	-0.004	-0.003	-0.002	0.000	0.001	0.003	0.004	0.005
4	-0.005	-0.004	-0.003	-0.002	-0.001	0.001	0.002	0.004	0.005	0.006
6	-0.004	-0.003	-0.002	-0.001	0.000	0.002	0.003	0.004	0.005	0.006
8	-0.004	-0.003	-0.001	0.000	0.001	0.003	0.004	0.005	0.006	0.007
10	-0.003	-0.002	0.000	0.001	0.002	0.004	0.005	0.006	0.007	0.007
12	-0.002	-0.001	0.000	0.002	0.003	0.005	0.005	0.006	0.007	0.008
14	-0.001	0.000	0.001	0.002	0.004	0.005	0.006	0.007	0.008	0.008
16	0.000	0.001	0.002	0.003	0.005	0.006	0.007	0.007	0.008	0.008

Figure F.12: Cantilever Aircraft Tailplane Trim Drag

Horizontal tail AoA	-12	-10	-8	-6	-4	-2	0	2	4	6
Elevator deflection										
-16	0.25	0.242	0.224	0.214	0.199	0.192	0.186	0.179	0.173	0.129
-14	0.242	0.223	0.214	0.198	0.188	0.173	0.171	0.161	0.158	0.088
-12	0.229	0.213	0.197	0.181	0.172	0.161	0.155	0.145	0.097	0.123
-10	0.209	0.198	0.181	0.169	0.155	0.144	0.135	0.127	0.096	0.109
-8	0.192	0.181	0.163	0.144	0.137	0.121	0.114	0.107	0.091	0.091
-6	0.173	0.159	0.143	0.128	0.111	0.104	0.091	0.062	0.076	0.057
-4	0.161	0.138	0.122	0.106	0.092	0.076	0.068	0.034	0.042	0.024
-2	0.141	0.118	0.102	0.08	0.069	0.05	0.038	0.022	0.006	-0.011
0	0.109	0.089	0.065	0.046	0.029	0.016	0	-0.015	-0.029	-0.046
2	0.076	0.05	0.032	0.012	-0.006	-0.019	-0.038	-0.049	-0.064	-0.079
4	0.037	0.015	-0.007	-0.026	-0.037	-0.034	-0.071	-0.081	-0.093	-0.108
6	0.001	-0.02	-0.042	-0.057	-0.072	-0.062	-0.097	-0.106	-0.118	-0.125
8	-0.035	-0.058	-0.069	-0.083	-0.067	-0.107	-0.12	-0.122	-0.133	-0.155
10	-0.065	-0.081	-0.1	-0.106	-0.096	-0.127	-0.142	-0.151	-0.162	-0.176
12	-0.096	-0.112	-0.12	-0.113	-0.097	-0.155	-0.152	-0.171	-0.182	-0.182
14	-0.138	-0.146	-0.152	-0.087	-0.161	-0.162	-0.167	-0.178	-0.189	-0.201
16	-0.155	-0.128	-0.145	-0.174	-0.176	-0.179	-0.183	-0.194	-0.204	-0.216

Figure F.13: Cantilever Aircraft Tailplane Pitching Moment

Horizontal tail AoA	-12	-10	-8	-6	-4	-2	0	2	4	6
Elevator deflection										
-16	-0.189	-0.196	-0.193	-0.191	-0.181	-0.174	-0.157	-0.128	-0.104	-0.067
-14	-0.192	-0.188	-0.191	-0.182	-0.176	-0.159	-0.144	-0.113	-0.087	-0.053
-12	-0.188	-0.187	-0.182	-0.172	-0.165	-0.150	-0.129	-0.098	-0.060	-0.036
-10	-0.181	-0.182	-0.177	-0.167	-0.154	-0.136	-0.112	-0.080	-0.046	-0.018
-8	-0.174	-0.175	-0.166	-0.152	-0.141	-0.116	-0.092	-0.062	-0.028	0.002
-6	-0.168	-0.165	-0.157	-0.143	-0.122	-0.103	-0.073	-0.036	-0.008	0.025
-4	-0.160	-0.156	-0.146	-0.130	-0.110	-0.081	-0.053	-0.018	0.018	0.047
-2	-0.156	-0.142	-0.133	-0.109	-0.091	-0.059	-0.028	0.010	0.042	0.069
0	-0.142	-0.132	-0.110	-0.089	-0.065	-0.038	0.000	0.035	0.065	0.089
2	-0.131	-0.110	-0.098	-0.074	-0.045	-0.010	0.030	0.059	0.086	0.109
4	-0.109	-0.092	-0.077	-0.051	-0.018	0.018	0.055	0.081	0.105	0.125
6	-0.093	-0.075	-0.056	-0.026	0.009	0.036	0.076	0.099	0.121	0.139
8	-0.082	-0.060	-0.030	-0.002	0.028	0.062	0.095	0.116	0.136	0.152
10	-0.058	-0.035	-0.010	0.018	0.046	0.080	0.113	0.132	0.150	0.163
12	-0.040	-0.016	0.009	0.040	0.060	0.104	0.122	0.146	0.161	0.172
14	-0.025	0.001	0.028	0.053	0.088	0.113	0.136	0.159	0.172	0.181
16	-0.006	0.018	0.043	0.077	0.105	0.129	0.149	0.169	0.181	0.188

Figure F.14: Cantilever Aircraft Tailplane Pitching Moment

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	503913	473861	447003	423160	402152	383799	367922	354342	342879	333353	325587	319399	314466
0.9529	2000	489535	460341	434249	411086	390677	372848	357424	344231	333096	323842	316297	310286	305494
0.9075	4000	475405	447053	421715	399220	379400	362086	347107	334295	323481	314494	307167	301329	296676
0.8217	6000	447902	421191	397319	376125	357452	341139	327027	314956	304767	296301	289397	283897	279513
0.7812	8000	434523	408610	385450	364890	346774	330949	317258	305548	295664	287450	280753	275417	271164
0.7423	10000	421408	396277	373817	353877	336308	320960	307683	296326	286740	278774	272279	267104	262979
0.7048	12000	408503	384141	362368	343040	326009	311131	298260	287251	277959	270237	263941	258924	254926
0.6689	14000	395888	372278	351178	332446	315941	301523	289050	278381	269375	261892	255790	250929	247053
0.6012	16000	371336	349191	329399	311829	296347	282823	271123	261116	252669	245650	239926	235367	231732
0.5694	18000	359423	337988	318832	301825	286840	273750	262425	252739	244563	237769	232229	227816	224297
0.5389	20000	347744	327006	308472	292018	277520	264855	253899	244527	236617	230043	224684	220414	217010
0.5096	22000	336274	316220	298297	282385	268366	256119	245524	236461	228812	222455	217272	213143	209851
0.4549	24000	314127	295393	278651	263788	250691	239251	229353	220888	213742	207804	202963	199105	196030
0.4292	26000	303355	285264	269096	254742	242095	231046	221489	213313	206413	200678	196003	192278	189308
0.4047	28000	292843	275379	259771	245915	233706	223040	213814	205922	199260	193725	189211	185615	182749
0.3813	30000	282563	265712	250652	237282	225502	215211	206308	198693	192265	186924	182569	179099	176333
0.3589	31000	272483	256233	241710	228817	217457	207533	198948	191605	185406	180256	176056	172710	170043

Figure F.15: Cantilever Aircraft Engine Maximum Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	438185	412053	388699	367965	349697	333738	319932	308123	298155	289873	283119	277738	273449
0.9529	2000	425683	400296	377608	357466	339719	324216	310804	299332	289648	281602	275041	269814	265647
0.9075	4000	413395	388742	366709	347148	329913	314857	301832	290691	281288	273473	267102	262025	257979
0.8217	6000	389480	366253	345494	327066	310828	296643	284371	273875	265015	257653	251650	246867	243055
0.7812	8000	377846	355313	335174	317296	301543	287782	275877	265694	257099	249957	244133	239493	235794
0.7423	10000	366442	344589	325058	307719	292442	279096	267550	257675	249339	242412	236764	232265	228678
0.7048	12000	355220	334036	315103	298295	283486	270549	259357	249784	241703	234989	229514	225152	221674
0.6689	14000	344250	323720	305372	289084	274732	262194	251347	242070	234239	227732	222426	218199	214829
0.6012	16000	322901	303644	286434	271155	257693	245933	235759	227057	219712	213609	208632	204667	201506
0.5694	18000	312542	293903	277245	262456	249426	238043	228196	219773	212664	206756	201939	198101	195041
0.5389	20000	302387	284353	268237	253929	241322	230309	220781	212632	205754	200038	195377	191664	188704
0.5096	22000	292412	274974	259388	245553	233362	222712	213499	205618	198967	193439	188932	185342	182479
0.4549	24000	273154	256864	242305	229380	217992	208044	199438	192076	185863	180699	176489	173135	170461
0.4292	26000	263787	248056	233996	221515	210517	200910	192599	185490	179489	174503	170437	167198	164616
0.4047	28000	254646	239460	225888	213839	203223	193948	185925	179062	173270	168456	164531	161404	158912
0.3813	30000	245707	231054	217958	206332	196088	187140	179398	172776	167187	162543	158756	155738	153333
0.3589	31000	236942	222811	210183	198972	189093	180464	172998	166613	161223	156744	153092	150183	147863

Figure F.16: Cantilever Aircraft Engine Takeoff Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	100157	94184	88845	84106	79931	76283	73127	70428	68150	66257	64713	63483	62503
0.9529	2000	97299	91496	86310	81707	77650	74106	71041	68419	66205	64366	62866	61672	60719
0.9075	4000	94490	88855	83819	79348	75409	71967	68990	66444	64294	62508	61052	59892	58967
0.8217	6000	89024	83715	78970	74758	71046	67804	64999	62600	60575	58892	57520	56427	55555
0.7812	8000	86365	81214	76611	72525	68924	65779	63058	60730	58765	57133	55802	54741	53896
0.7423	10000	83758	78763	74299	70336	66844	63793	61154	58897	56992	55409	54118	53089	52269
0.7048	12000	81193	76351	72024	68182	64797	61840	59281	57093	55246	53712	52460	51463	50668
0.6689	14000	78686	73993	69799	66076	62796	59930	57451	55330	53540	52053	50840	49874	49104
0.6012	16000	73806	69404	65471	61978	58901	56213	53888	51899	50220	48825	47687	46781	46058
0.5694	18000	71438	67178	63370	59990	57012	54410	52159	50234	48609	47258	46157	45280	44581
0.5389	20000	69117	64995	61311	58041	55159	52642	50464	48602	47029	45723	44658	43809	43132
0.5096	22000	66837	62851	59289	56126	53340	50906	48800	46998	45478	44215	43185	42364	41710
0.4549	24000	62435	58712	55384	52430	49827	47553	45586	43903	42483	41303	40340	39574	38963
0.4292	26000	60294	56698	53485	50632	48118	45922	44023	42398	41026	39886	38957	38217	37627
0.4047	28000	58205	54734	51632	48877	46451	44331	42497	40929	39604	38504	37607	36892	36323
0.3813	30000	56162	52812	49819	47162	44820	42775	41005	39492	38214	37153	36287	35597	35048
0.3589	31000	54158	50928	48042	45479	43221	41249	39542	38083	36851	35827	34992	34327	33797

Figure F.17: Cantilever Aircraft Engine Cruise Thrust as function of Altitude and Mach Number

	Mach No	0.06	0.12	0.18	0.24	0.30	0.36	0.42	0.48	0.55	0.61	0.67	0.73	0.79
Relative Density	Altitude													
1	0	80125	75347	71076	67285	63945	61026	58502	56343	54520	53005	51770	50786	50002
0.9529	2000	77839	73197	69048	65365	62120	59285	56833	54735	52964	51493	50293	49337	48575
0.9075	4000	75592	71084	67055	63479	60327	57574	55192	53155	51435	50007	48841	47913	47173
0.8217	6000	71219	66972	63176	59806	56837	54243	51999	50080	48460	47114	46016	45141	44444
0.7812	8000	69092	64971	61289	58020	55139	52623	50446	48584	47012	45706	44641	43793	43117
0.7423	10000	67007	63010	59439	56269	53475	51035	48923	47118	45593	44327	43294	42471	41815
0.7048	12000	64954	61081	57619	54545	51837	49472	47425	45675	44197	42969	41968	41171	40535
0.6689	14000	62949	59195	55840	52861	50237	47944	45961	44264	42832	41642	40672	39899	39283
0.6012	16000	59045	55523	52376	49583	47121	44971	43110	41519	40176	39060	38150	37425	36847
0.5694	18000	57150	53742	50696	47992	45609	43528	41727	40187	38887	37807	36926	36224	35665
0.5389	20000	55294	51996	49049	46433	44127	42114	40371	38881	37624	36578	35726	35047	34506
0.5096	22000	53470	50281	47431	44901	42672	40724	39040	37599	36382	35372	34548	33891	33368
0.4549	24000	49948	46969	44307	41944	39861	38042	36469	35123	33986	33042	32272	31659	31170
0.4292	26000	48235	45359	42788	40506	38495	36738	35218	33918	32821	31909	31166	30573	30101
0.4047	28000	46564	43787	41305	39102	37161	35465	33998	32743	31684	30803	30086	29514	29058
0.3813	30000	44929	42250	39855	37729	35856	34220	32804	31593	30571	29722	29030	28478	28038
0.3589	31000	43326	40743	38433	36383	34577	32999	31634	30466	29481	28662	27994	27462	27038

Figure F.18: Cantilever Aircraft Engine Cruise Thrust (80%) as function of Altitude and Mach Number

Stability and Control Graphs

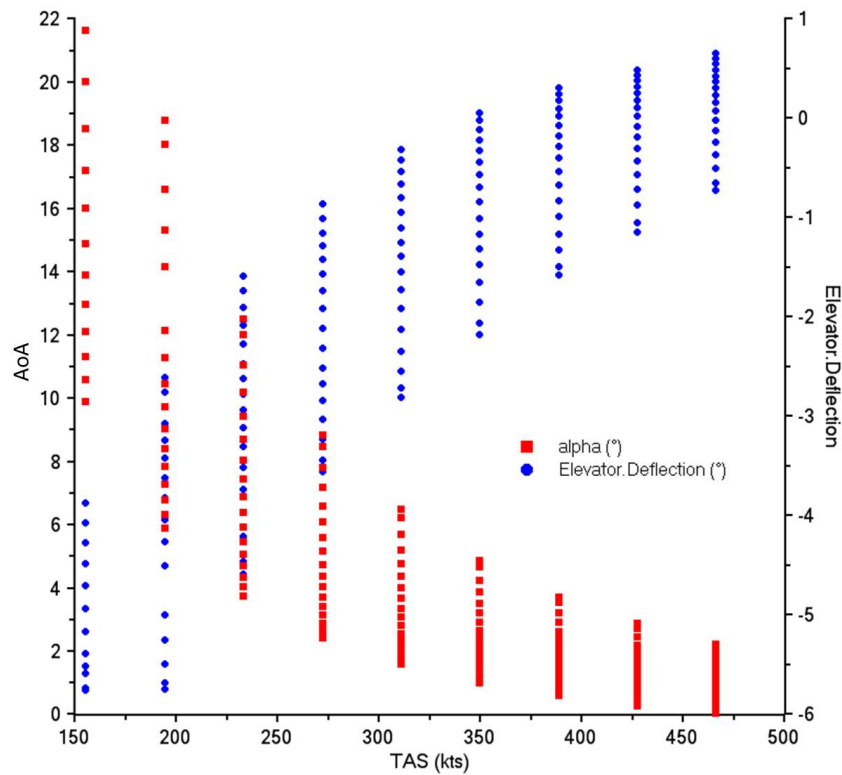


Figure G.1: AoA and Elevator Deflection to Trim - 66% Payload

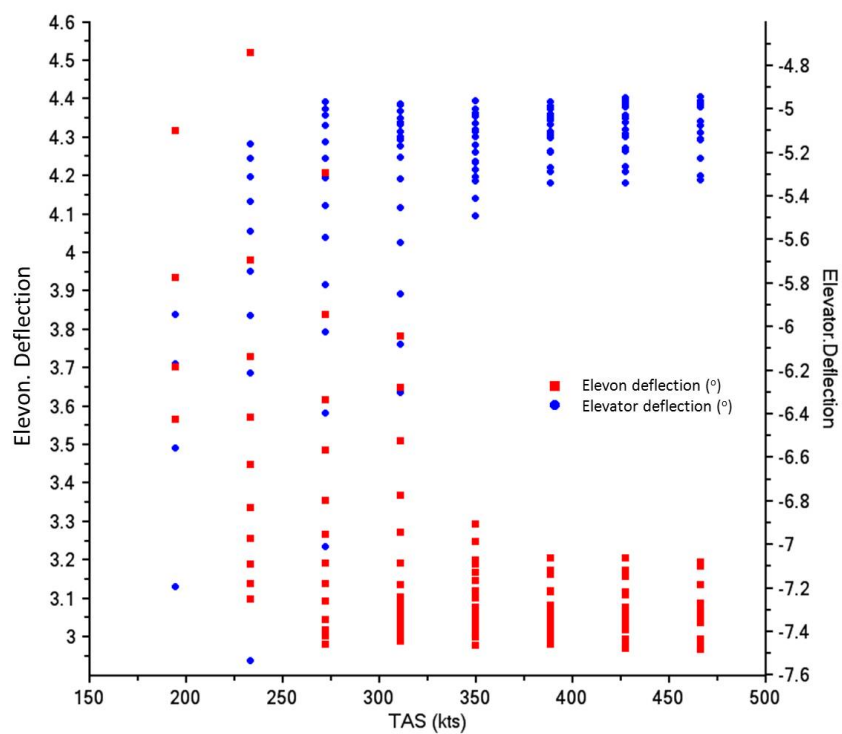


Figure G.4: Box Wing Elevon and Elevator Deflection to Trim - 33% payload

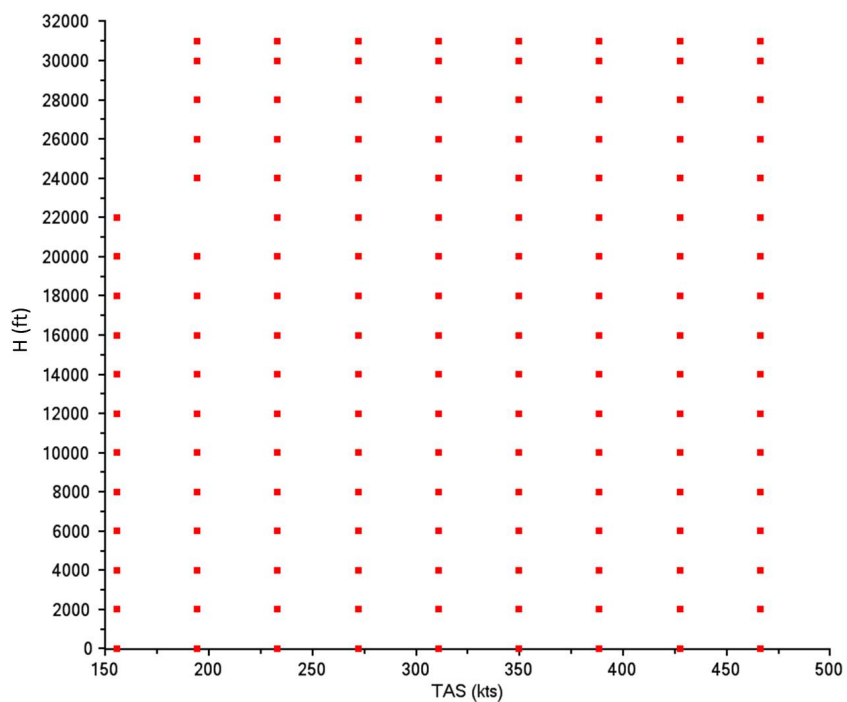


Figure G.5: Flight Envelope Achievable - 66% Payload

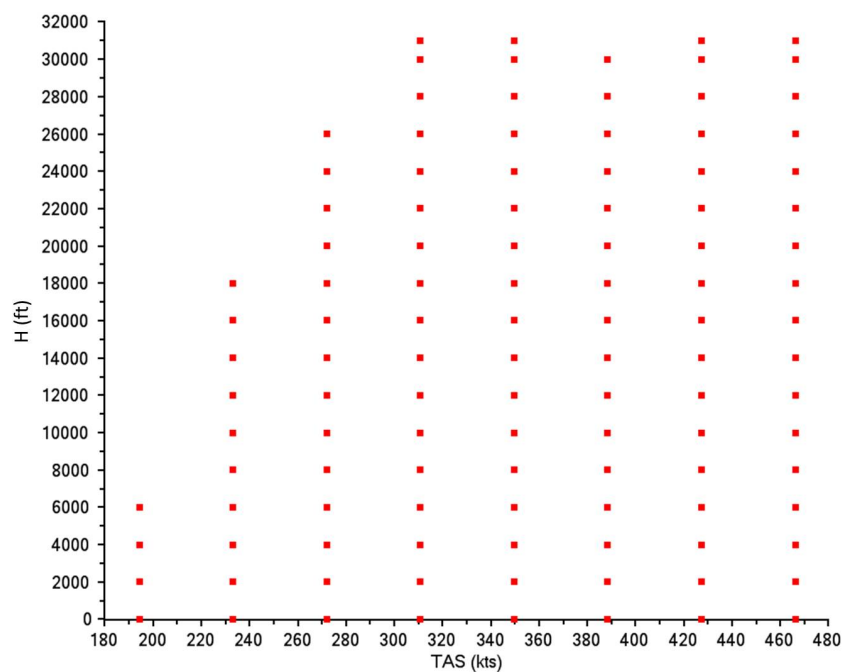


Figure G.6: Box Wing Flight Envelope Achievable - 66% payload

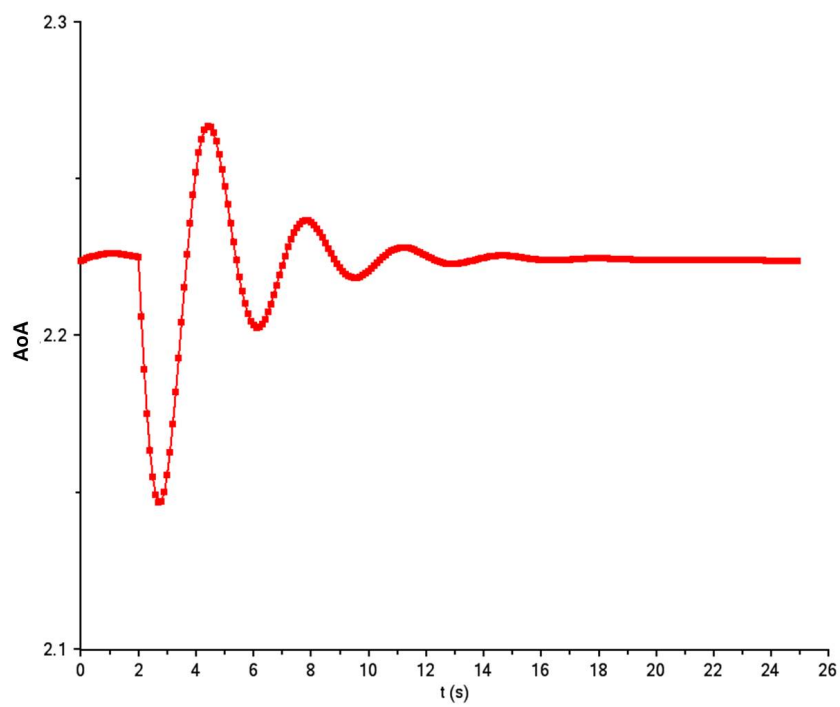


Figure G.7: Short Period Oscillation - 66% Payload

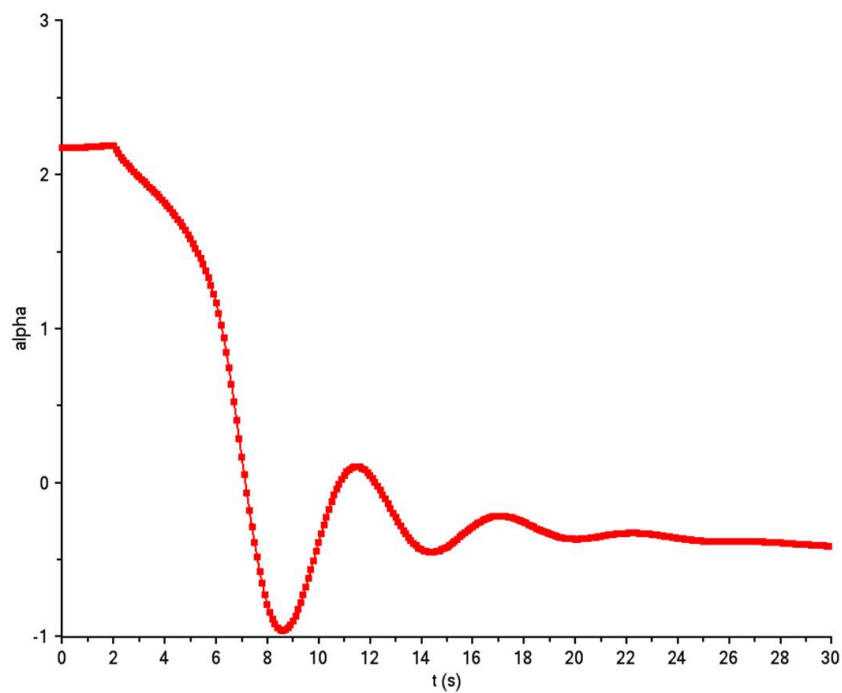


Figure G.8: Box Wing Short Period Oscillation - 66% payload

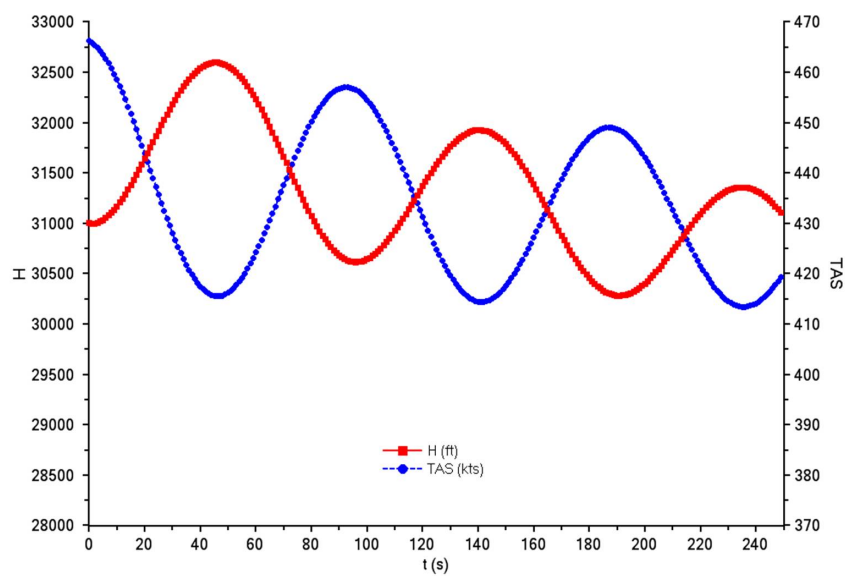


Figure G.9: Phugoid - 66% payload

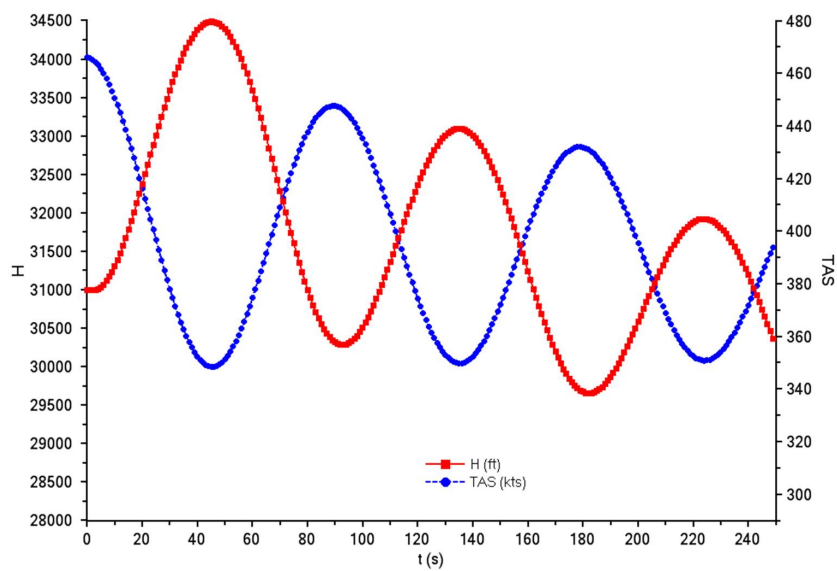


Figure G.10: Box Wing Phugoid - 66% payload