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Study of blended wing body design concept

Thesis Submitted in Partial Fulfillment of the Requirements for the Degree of Bachelor of Science. (BSc Honor)

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قال تعالى:

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﴿ وَابْتَغِ فِيمَا آتَاكَ اللَّهُ الدَّارَ الْآخِرَةَ * وَلَا تَنْسَ نَصِيبَكَ مِنَ الدُّنْيَا *
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Abstract

This thesis represents a simple study of the design concept of a new generation of civilian aircrafts called "BLENDED WING BODY" aircraft , which are meant to replace the current cylindrical figure aircrafts in the sooner future .

The search and study began with the conceptual design phase , taking both AIRBUS 380 & BOEING 747 & BOEING 777 as design references as they are all of the same weight and passenger capacity (500) .

The design process started by setting the requirements based on historical data mostly, following all the conceptual design stages till the 3D-VIEWS was obtained. Going through airfoil selection , mission , weight estimation to the performance analysis of the aircraft which is mentioned in details .

To conclude this study, this type of aircrafts has less fuel consumption and a better performance when compared to conventional aircrafts. However , the only problem that had to be dealt with was the airfoil selection which was solved at the drawing phase . with a better solutions proposed for any future work.

Acknowledgement

Foremost , we would like to express our sincere gratitude to our advisor Mr.Abdelsameea for the continuous support of our thesis study and research , for his patience , motivation , enthusiasm , and immense knowledge . his guidance helped us in all the time of research and writing of this thesis . we could not have imagined having a better advisor and mentor for our study .

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Last but not least , we would like to thank whoever supported our idea and encourage us to keep working despite the harden ships.

We are forever grateful .

Dedication

Every challenging work needs self-efforts as well as guidance of elders especially those who are very close to our hearts.

Our humble effort , we dedicate to our loving families.

Mother , father , brothers & sisters

Whose affection , love , encouragement and prays of day and night makes us able to get such success and honor.

Along with all the hard working and respected teachers.

Contents

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List of Symbols

$(\frac{L}{D})_{max}$	Maximum lift to drag ratio
C	Specific fuel consumption
R	Range
$(C_L)max$	Maximum lift coefficient
V_{stall}	Stalling velocity
V_f	Flare velocity
R	Flight path radius
g	Acceleration of gravity
θ_a	Approach angle
h_f	Flare height
S_a	Approach distance
S_f	Flare distance
S_g	Ground roll
μ_r	Coefficient of the rolling friction
S	Wing area
ρ	Density
θ_{OB}	Flight path angle
S_a	Airborne distance
C_{fe}	Skin friction coefficient
S_{wet}	Wetted area
$C_{D,0}$	Zero lift drag coefficient
e	Oswald factor
k	Drag due to lift coefficient
AR	Aspect ratio
$({}^{R}/{}_{C})_{max}$	Maximum rate of climb
b	Wing span
λ	Taper ratio
Ć	Mean aerodynamic chord
C_r	Root chord
C_t	Tip chord
x	center of gravity location
$\frac{W}{S}$	Wing loading
$\frac{T}{W}$	Thrust to the weight ratio
W_0	Takeoff gross weight
W_e	Empty weight
W_f	Fuel weight

Chapter 1 : INTRODUCTION

1.1 Overview

A blended wing body (BWB) is a fixed wing aircraft having no clear dividing line between the wing and the main body of the craft . The form is composed of distinct wing and body structures though the wings are smoothly blended into the body unlike a flying wing which has no distinct fuselage .[1]

The BWB airframe merges efficient high-lift wings with a wide airfoil-shaped body, allowing the entire aircraft to generate lift and minimize drag .

The streamlined shape between fuselage and wing intersections reduces interference drag, reduces wetted surface area that reduces friction drag while the slow evolution of fuselage-to-wing thickness by careful design may suggest that more volume can be stored inside the BWB aircraft, hence, increases payload and fuel capacity[6,7] .

The BWB has several distinct advantages over the conventional tube aircraft. Some of these advantages are outlined below:

1-Higher fuel efficiency:

Initial testing of the BWB aircraft has indicated that it can have up to a 27% reduction in fuel burn during flight [8].

2-Higher payload capacity:

Due to the blended nature of the fuselage, the fuselage is no longer distributed along the centerline of the aircraft. As a result, the fuselage is more ―spread out,‖ allowing for greater volume and a larger payload capacity [8].

3. Lower takeoff weight:

Early design concepts have determined that the BWB can have up to a 15% reduction of take-off weight when compared to the conventional baseline [8].

4. Lower wetted surface area:

The compact design results in a total wetted difference of 14,300 ft2, a 33% reduction in wetted surface area. This difference implies a substantial improvement in aerodynamic efficiency [8].

5. Commonality:

One of the greatest advantages of the BWB is *commonality* of *size* and of *application* [9]. Firstly, the commonality of the components of the airplane will allow it the payload of the airplane to be varied at little cost. For the 250, 350, and 450 – passenger capacity of the BWB, many components are interchangeable. This interchangeability serves to drive down the cost of the aircraft. Secondly, commonality of function allows the BWB to be used in many applications, both military and civilian. The BWB can be modified to be used as a freighter, troop transport, tanker, and stand-off bomber in addition to its function as a commercial airliner.

1.2 PROBLEM STATEMENT

This project represents a study of the concept and performance of a blended wing body , going through conceptual design phase

1.3 Aim and Objective

1.3.1 Aim

The primary aim is to create a design of a blended wing body , with the capacity of 500 passenger based on Airbus 380 specifications

1.3.2 Objectives

1. achieve aircraft final configuration
2. performane analysis
3. aircraft initial sizing
4. structure analysis

1.4 METHODOLOGY

Firstly, we start with collecting data from differentsources ,mainly web. Then setting the requirements of the aircraft using historical data from available similar aircrafts as a reference .Secondly , we get into the conceptual design phase where we obtain the configuration of the aircraft using (CATIA V5) . followed by performance analysis using mathematical equations.

1.5 OUTLINES

Chapter one: introduction

Chapter two: literature review

Chapter three: conceptual design

Chapter four : results and discussion

Chapter five: recommendations and future work

Chapter 2 : literature Review

2.1 Difference between a blended wing body and a flying wing

- ☐ Flying wing designs are defined as having two separate bodies and only a single wing, though there may be structures protruding from the wing.

- ☐ Blended wing/body aircraft have a flattened and airfoil shaped body, which produces most of the lift to keep itself aloft, and distinct and separate wing structures, though the wings are smoothly blended in with the body.

In recent years unconventional aircraft configurations, such as Blended-Wing-Body (BWB) aircraft, are being investigated and researched with the aim to develop more efficient aircraft configurations, in particular for very large transport aircraft that are more efficient and environmentally-friendly. The BWB configuration designates an alternative aircraft configuration where the wing and fuselage are integrated which results essentially in a hybrid flying wing shape.

The first example of a BWB design was researched at the Lockhead Company in the United States ofAmerica in 1917. The Junkers G. 38, the largest land plane in the world at the time, was produced in 1929 for LuftHansa (present day; Lufthansa). Since 1939 Northrop Aircraft Inc. (USA), currently Northrop Grumman Corporation and the Horten brothers (Germany) investigated and developed BWB aircraft for military purposes. At present, the major aircraft industries and several universities has been researching the BWB concept aircraft for civil and military activities, although the BWB design concept has not been adapted for civil transport yet. The B-2 Spirit, (produced by the Northrop Corporation) has been used in military service since the late 1980s. The BWB design seems to show greater potential for very large passenger transport aircraft. A NASA BWB research team found an 800 passenger BWB concept consumed 27 percent less fuel per passenger per flight operation than an equivalent conventional configuration (Leiebeck 2005).

A BWB configuration has superior in flight performance due to a higher Lift-to-Drag (L/D) ratio, and could improve upon existing conventional aircraft, in the areas of noise emission, fuel consumption and Direct Operation Cost (DOC) on service. However, a BWB configuration needs to employ a new structural system for passenger safety procedures, such as passenger ingress/egress.

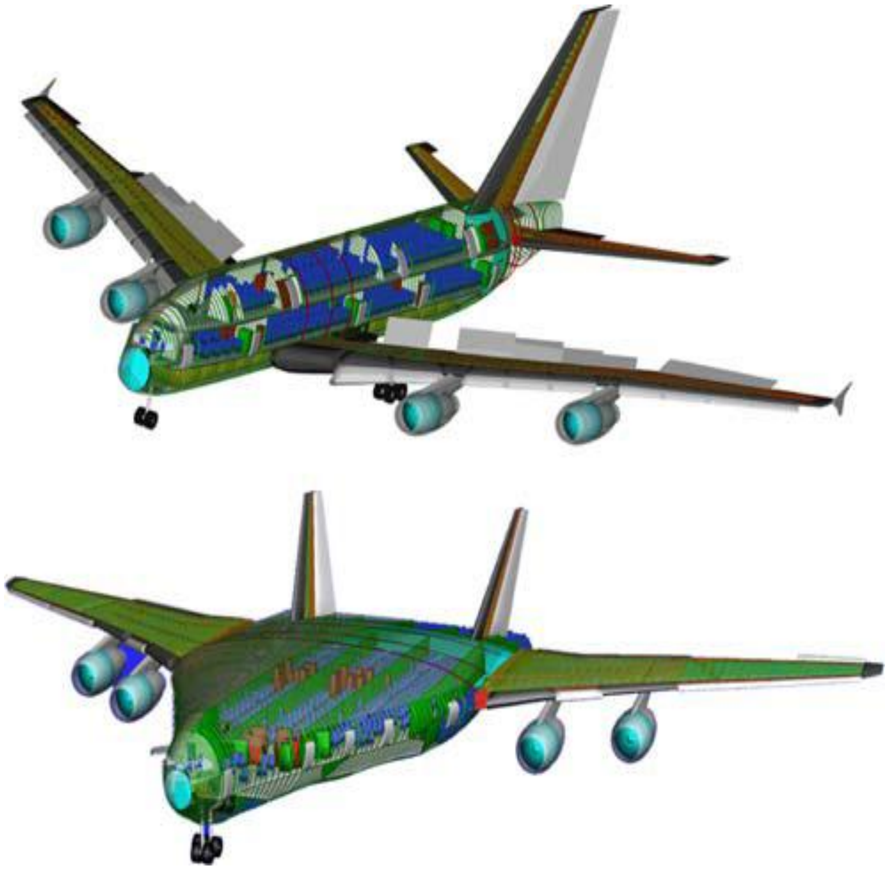

Figure 2.1

Aircraft technologies that could give greater performance include a large improvement in Lift-to-Drag ratio of a wing coupled to evolutionary improvement in composite structure and engines, such as Blended Wing Body aircraft configuration. This next generation airlifter has been researched with a high L/D ratio wing configuration design, engineered materials, composite fabrication and fastening,

and next generation material for airframe and skin. A Blended-Wing-Body (BWB) design approach is to maximise overall efficiency by integrated the propulsion systems, wings, and the body into a single lifting surface. This BWB configuration is a new concept in aircraft design which expects to offer great potential to substantially reduce operating costs while improving an aerodynamic performance and flexibility for both passenger and cargo mission.

In the United States, Sir W. G. Armstrong Whitworth Aircraft Ltd., designed the Armstrong Whitworth A.W. 52 (Fig. 2.1) in 1947 (British Aircraft 2005), and General

Dynamics/McDonnell Douglas was also selected to develop a subsonic twin jet carrier, A-12 Avenger II, based on Advanced Tactical Aircraft concept (ATA) for attack at night or in bad weather in 1990 (GloablSecurity.org 2005).

In Japan there were several Flying-Wing concept aircraft, such as the HK 1 (Fig 2.2) which was the first Japanese tailless aircraft produced by the Ito Aircraft Laboratory in 1939 (BWB World 2005). Since the first test flight the HK 1 had been flown at 116 times, and the chiefengineer Mr. Kimura reported that the HK 1 has a quiet, stable flight control in test flight at 1,000 meters altitude.

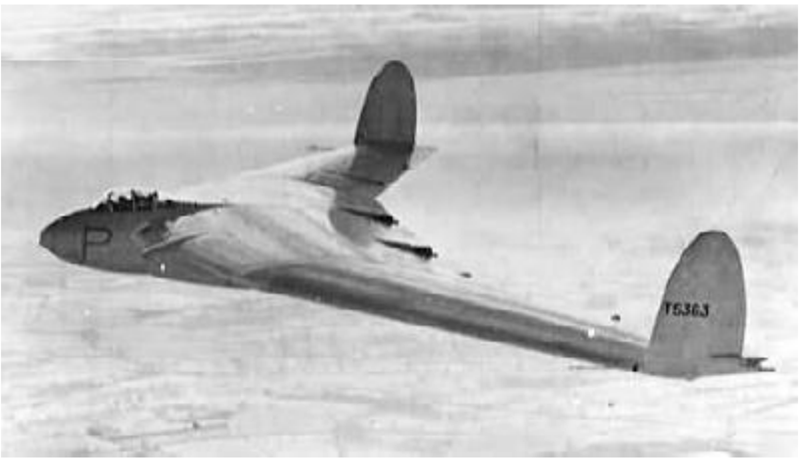

Figure 2.2 Armstrong Whitworth A.W. 52

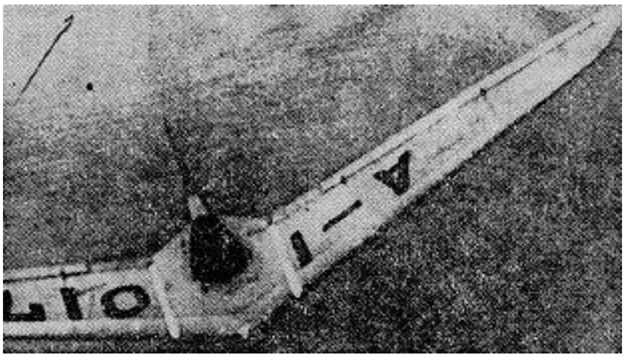

Figure 2.3 Kayaba HK 1

In more recent years major aeronautical industries and universities have been researching and developing performance of BWB configuration for commercial aircraft. In regards to the research project at Cranfield College of Aeronautics, the preliminary design project of the Blended Wing Body Airliner is currently at the cutting edge of aircraft design technology exploring and evaluating a new configuration. This research has discovered a great deal of advantages and these concepts can be summarised as Fig 2.3.

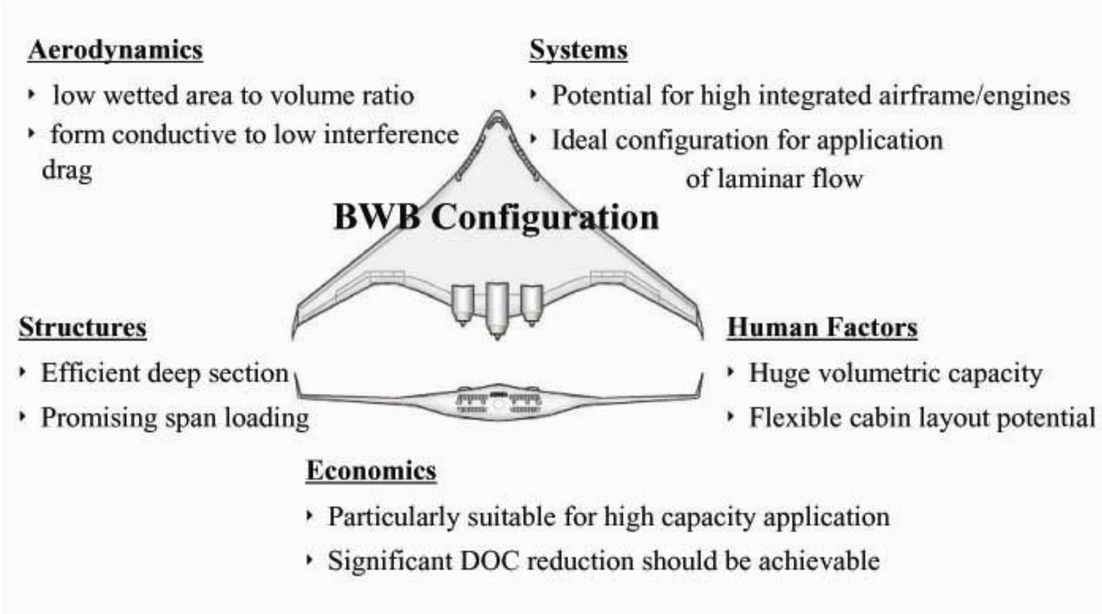

Figure 2.4 Features of BWB Aircraft Configuration

In 1991, the NASA Langley Research Centre built a model of a BWB aircraft which had three engine nacelles on the aft of the top surface. Regarding the nose emission of this BWB aircraft, Dr. Lorenzo noticed that this aircraft model could reduce noise. The noise radiated downward was reduced by 20 dB to 25 dB overall in the full scale frequencies from 2,000 to 4,000 Hz, decreasing to 10 dB or less at the lower frequencies (Sandilands 2002).

NASA Langley Research Centre has stated that a BWB configuration will be more of a 600 than an 800 passenger airliner. Concerning this BWB aircraft, aeroelastic deflection will be severe for the wing span and will be counteracted by active surface as well as for verticals to provide a directional stability, control and to act as winglets to increase the effective aspect ratio (Guynn et al. 2004).

The leader of the aircraft industry, the Boeing Company, has announced that a BWB aircraft would climb an extremely steep angle, even compared to the gun-ho steep climb out experienced in the successful passenger flights today. In regards to the comparison between BWB configuration and conventional aircraft which have the same number of passengers and range for particularly large airliners, the BWB would be lighter and have a higher Lift-to-Drag (L/D) ratio and less fuel burn. For

example, the BWB-450 which has been designed by the McDonnell Douglass team since 1988 would use 32 percent less fuel per seat and be 18 per cent lighter at its maximum Take-Off Gross Weight (TOGW) if both jets carried 480 passengers for an 8,700 nautical mile flight. In reference to the structural analysis of BWB aircraft, the configuration would require 30 percent fewer parts than conventional aircraft, because there are no complex wing-fuselage and fuselage-empennage joints

(Sandilands 2002).

In regards to the high-lift-wing design for a Megaliner aircraft of Airbus A380-800 (Reckzen 2002), powered high-lift systems (e.g. externally blown flaps) of the Airbus Company showed an impressive maximum lift potential beyond the performance of the familiar conventional high-lift systems. The high-lift performance aircraft, such as the A380 prototype, contributed better benefits than conventional aircraft which can be summarised as;

- 5 percent in maximum lift leads to 12-15 percent increase of payload,

- 5 percent of take-off L/D leads to 20 percent increase of payload,

- 5 percent of maximum lift in landing configuration leads to 25 percent increase of payload.

To conclude, the BWB aircraft configuration, synthetically, has the ability to provide a great number of benefits through its structural concepts, such as its aerodynamically low interface drag, high lift-to-drag ratio, structurally favourable span loading, and the reduction of green house emissions.

Figure 2.5 Boeing Joined-Wing Concept Configuration (Steinke 2001)

In regards to its control stability, the stable all-wing configuration (Fig 2.4) is difficult to trim without resorting to download at the wingtip which increases drag. The BWB concept design relies on advanced flight control systems to provide stable flight control allowing the centre-of-gravity to move the aft without trim problems. Furthermore improvement of the concept design is realised through use of boundary layer integration in the engines. This engine installation which is on the aft of the body allows the engines to scavenge a sizable portion of the boundary air of the shape reducing the inlet ram drag and increasing efficiency. With the body shape of the BWB concept, the BWB configuration is predicted to be a ‘clean’ and more ‘environment friendly’ from of transport.

Cranfield College of Aeronautics in the UK is an advocate of the flying wing or BWB as offering a major step forward in overall efficiency. Moreover, he states that this future vision of aircraft design is very impressive as he said (Birch 2001), By using nuclear fuel to power what is essentially a closed-cycle stream engine driving propellers, there would be no atmospheric emissions to cause concern. Therefore, there would probably be some degradation in airliner cruising speed - about Mach 0.7 would be typical - but the efficiency of the aircraft, without the need to carry an enormously heavy fuel load ontake-off, would be very high. Cranfield College said that while the researchers fully understand that public and political unease about nuclear-powered aircraft would be considerable, and nevertheless feel that the use of nuclear power should be considered as a serious alternative aviation fuel. Considerable interest has been raised by the fact that the BWB layout may confer substantial overall advantages when applied to a transport aircraft in the ultra-high-capacity category. The most famous BWB design of the Cranfield College of Aeronautics is the College of Aeronautics BW-98 projectillustrated in Fig. 2.6 and Fig 2.7 (Howe 2001 & Smith 2000). This Cranfield baseline BWB configuration is similar to the Boeing concept in configuration, and currently represents the only UK National project of its scale.

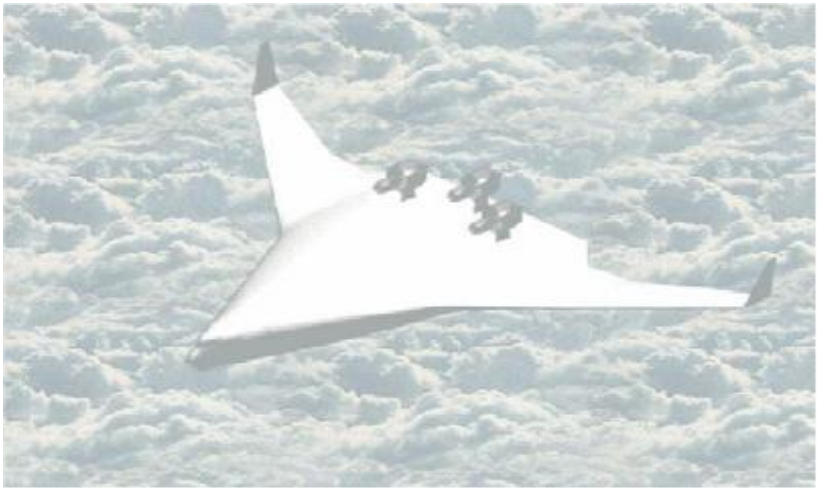

Figure 2.6

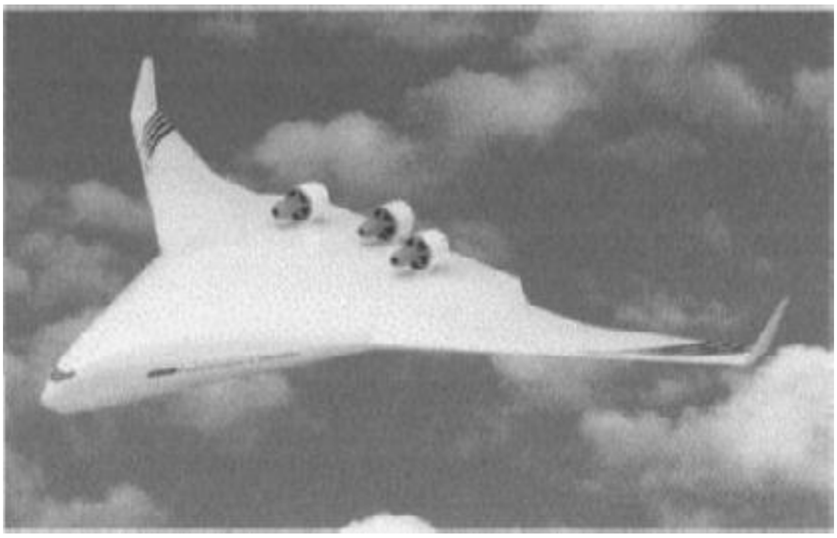

Figure 2.7 Cranfield BW-98 BWB Study (up: Smith 2000, down: Howe 2001)

2.2 Negative factors of a BWB Configuration Design

Unconventional aircraft configuration with the BWB design have been predicted to pose design challenges in this new class modelling to achieve the BWB projections. The majority of issues involved in the BWB design stage involve the structural capabilities based on the physics analysis of Finite Element Method (FEM), aerodynamic panel-method, and drag and weight prediction, and also a number of problems in aerodynamic performance of the configuration are not understood yet.

In regards to the structural analysis of the BWB concept, the stress level in the box type of pressurised fuselage configuration of the BWB flight vehicle is an order to magnitude higher, because internal pressure primarily results in blending stress instead of skin-membrane stress. Moreover, resulting deformation of aerodynamic surface significantly affects flight performance provided by the lifting body. For example, the pressurised composite conformal multi-lobe tanks of X-33 type space aircraft also suffered from the similar problem (Mukhopadhyay 2005).

Another problem related to the human factor in the wide cabin design model is how to install the windows. The passenger compartment goes into the wing structure area, so it is difficult to set up the windows on the wing surface. Also the outside of the passenger area will be located tank running out into the wings. To solve this problem, a multi-functional liquid crystal display (LCD) screen on the seat for the rear passengers and several windows for the front passengers will be installed.

In regards to the aerodynamic effects of the all delta wing concept, negative aeroelasticbehaviours in cruise due to elastic deformations is considered to change in wing twist, aileron reversal of flatter identified which will be overcome by considerable changes in structural arrangement or mass distribution, resulting in weight penalty or unacceptable limitations of the flight envelop. Moreover,

the relationship between the engine location and the flight operation is critical to solve the inlet and compressor problems with the turbulence flow off the rear of the wings, because the BWB concept design has the engines raised out of the boundary layer flow when the angle of attack is higher in flight.

important advantage of BWB's derives from the strategy of combining the fuselage

and the wings into one body: reduced structural weight. As illustrated in Figure 6, the distribution of loads spanwise across a traditional aircraft leads to most of the weight being carried in the center and nearly all of the lift coming from the wings. The wings must therefore be able to resist massive bending loads. However, in a blended-wing-body, the loads are more evenly distributed across the span (i.e. high weight sections have high lift and lighter portions of the aircraft generate less lift). This leads to smaller bending moments on the aircraft structure Consequentially, structural requirements of the aircraft are lessened and empty weight is reduced, further increasing efficiency.(36)

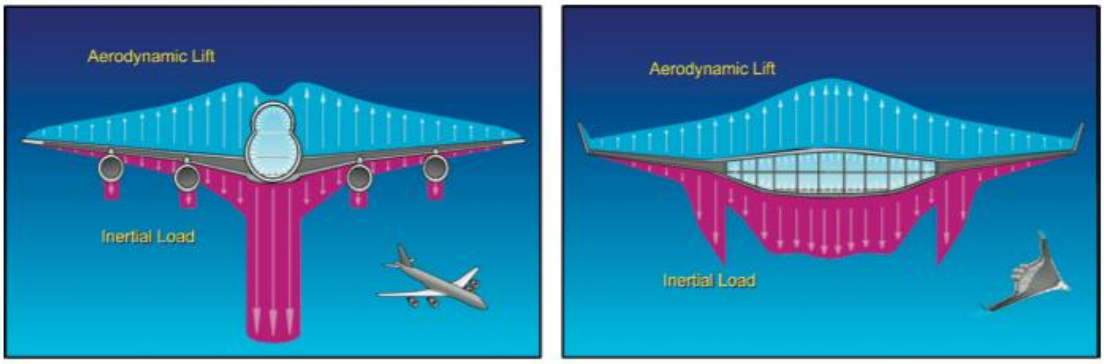

Figure 2.8 Distribution of Loads in a conventional aircraft vs. a blended-wing-body

2.3 Difficulties in Designing a BWB

here are two central reasons why a blended-wing-body aircraft is a technically challenging engineering project:

The first of these reasons is the difficulty associated with the manufacture of BWB's while the other problem stems mostly from aerodynamic design considerations. While these reasons appear daunting at first glance, sufficient research has been conducted so that all of these issues could be overcome. If aerospace companies want to push their product lines into the next era of commercial passenger transport, they need to move towards developing a blended-wing-body. One of the difficulties that arises when considering the design of a BWB is the complicated manufacturing process that is associated with a new airplane design. Conventional tube-and-wing design aircraft have two particular advantages. Because aircraft have been made this way for half a century, much is known about how to assemble efficiently the various components, large and small, of this type of airplane. In addition, the tube design of the fuselage makes assembly and customization quite literally a simple process of adding nearly identical cylindrical components. A blended-wing-body, by its nature, is a much more complicated structure to design, manufacture, and assemble. [9] Thankfully, research has been performed that has shown that many of the same principles that are applied to conventional designs today can be used in BWB design. Airplanes can be “molecularized” which means that its design can be separated into components that can be added and subtracted to allow for simple aircraft customization, as seen in Figure 9. This way the needs of each airline customer can be met without having to create individualized designs. This process is detailed and organized in a patent filed for and given to the Boeing Company in 2003. [9] Using this or similar methods, BWB design and manufacture could be as or more streamlined than the assembly lines of today's aircraft manufacturers. With manufacturing problems solved, the aerodynamic concerns remain the only impediments to the success of the blended-wing-body. At the forefront of these issues is the required capabilities that must be in place to create such a design. By its very nature, the BWB is a particularly integrated system and thus the process of design requires the study of many simultaneously self-dependent variables. [8] Conventional aircraft design requires a tremendous number of aerodynamic performance studies. Blended-wing-body design will require even more analysis and computational work. Luckily, one particular advantage that modern aircraft engineers have in their design toolkit is Computational Fluid Dynamics, or CFD. This technology uses the power of computers to simulate the aerodynamics of any aircraft, as shown in Figure 10. Already a great deal of research has been done to determine what are the best methods to use in the study of BWB's using CFD. [10] With CFD in hand, much of the work of aeronautical engineers is simplified to working with a digital model and making small adjustments to design based on theoretical intuition blended with careful analysis of simulation results. Utilization of this technology will reduce the difficulty of designing an integrated aircraft so that expertise of engineers can be focused on design and not on vast volumes of computation. Another difficulty that arises when designing a BWB are the limitations associated with stability and control of the aircraft. Conventional aircraft have long fuselages with tailplanes located at the rear to statically stabilize the aircraft and to give the control surfaces sufficient control power. The flying wing design of the blended-wing-body aircraft improves aerodynamic efficiency in part by doing away with a tailplane. Unfortunately, this makes the task of designing controls that can make the aircraft stable and safe much more difficult. Luckily, flying wing designs have been used before and some have been quite successful (e.g. the Lockheed B-2 bomber). Sufficient research into the control aspects of BWB design has been conducted to show that the creation of a safe blended-wing-body aircraft is quite plausible. [11] Fly-by-wire technology, which routes flight control commands from the pilot through a computer which in turn controls the movements of the aircraft, will prove invaluable in making this possible. This technology has been proven safe and reliable and is now used on many commercial aircraft that are flown today. Thus, controlling a BWB would be a challenging but very possible engineering feat. The final and most difficult aerodynamic challenge that remains is the difficulty of sculpting a passenger and cargo cabin into a flying wing. [8] Today wing and cabin design are essentially separate endeavors. However, because in a BWB the wing is the fuselage, the cabin and cargo areas must be adjusted to fit inside of an aerodynamic structure. As a result of this feature, several constraints come to the forefront of the design process. The first of these is the constraint that the wing not be too thick. Normally, airplane wings that travel at speeds approaching the speed of sound require thin wings, but BWB designs are inherently thick because the aircraft's payload is stored inside the wing. This problem is solved by shaping the wing in different ways so as to avoid the negative effects of Mach drag (the drag resulting from traveling near the speed of sound). Sweeping back the wings of the aircraft delays the onset of Mach drag and thus much research has been performed on the viability of using sweep to combat effectively Mach drag effects. [12] This technology could effectively solve the thickness problem while still maintaining the cabin size requirements.

The second limitation is more of an imposed constraint, but it is equally if not more important than all the others. Since the BWB will carry passengers in a non conventional way, its design must be able to meet cabin egress requirements set by aviation authorities. Many BWB cabin designs put passengers at further

distances from emergency exits than conventional aircraft, so there is the worry that people would be unable to escape a crashed aircraft within the ninetysecond-

time limit. However, with the use of simulation and with innovative design and emergency exit placement, it can be shown that BWB egress performance is comparable to that of a conventional aircraft. [6] Using tools like this, the design of the BWB cabin will be a challenge not without its complications and difficulties but one that is entirely within the reach of today's engineers. (37)

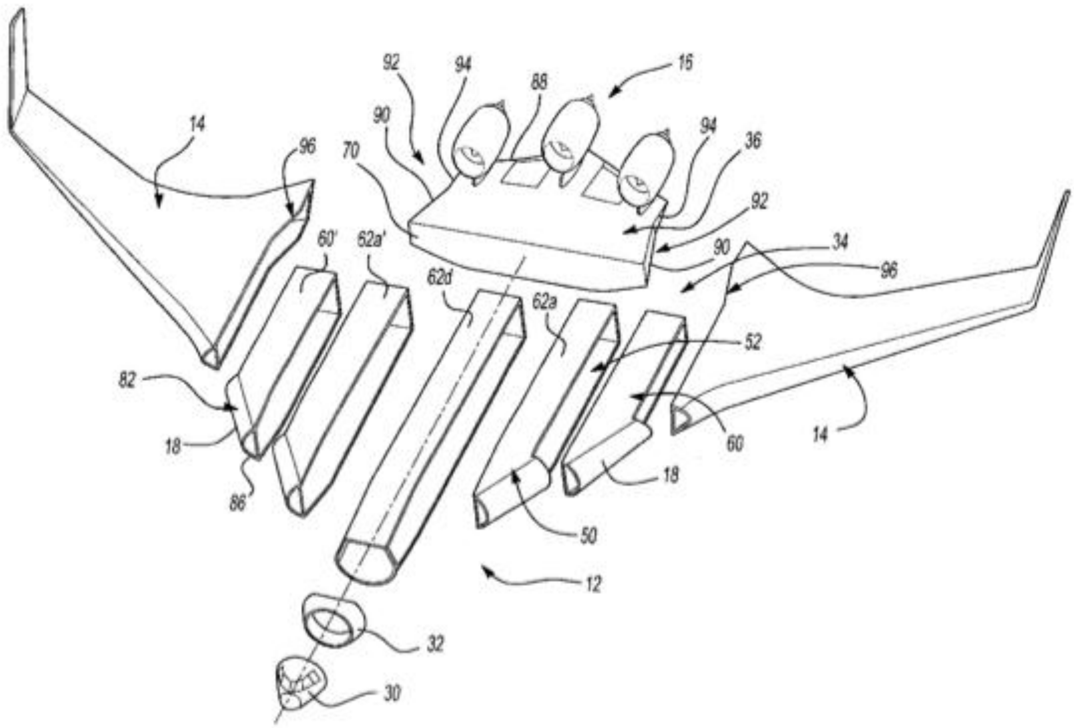

Figure2.9 Molecularization” of the BWB Design

2.4 Environmental benefits

increasing environmental concerns are a global issue faced by many industries, and the aerospace industry is no exception. While aviation is responsible for about 4.9% of global greenhouse emissions [27], growth in the industry is only expected to increase. For instance, between 2009 and 2029, Boeing predicts an annual growth of 5.3% in total world traffic flow [68]. Aviation is a culprit for more than just the expected growth [77]: _first, planes are delivering nitrogen oxides at higher altitudes and therefore, very efficiently contributing to the problem. Combined with the effects of cirrus clouds from contrails, the direct effect of carbon dioxide is tripled [65, 77]! Secondly, since not everyone is flying or can even afford to y, an inequality in responsibility exists. Finally, immediate alternatives to fossil fuels are not quite available: hydrogen could add to cirrus clouds; biofuels are promising but unlikely to replace standard jet fuel for several decades, leaving fuel efficiency improvement as one of the key options [77]. Improving fuel efficiency is not only important from an environmental perspective, but also from an economics perspective: with increasing fuel prices | up by almost 13% from about a year ago [22] | airlines will need more fuel efficient options given the predicted growth in the air traffic. The work in this thesis brings two ideas together | the unconventional blended-wing-body aircraft configuration and high-fidelity aerodynamic shape optimization | in the hopes of effectively contributing to the development of future solutions which are more fuel-efficient and environmentally-friendly.

2.6 Advantages of BWB

2.6.1 Aerodynamics

A key aspect of the BWB is its lift-generating centerbody | a gain over the cylindrical

fuselage of a conventional aircraft | which improves the aerodynamic performance by reducing the wing loading [46, 48, 63]. In addition, the decrease in wetted area, via a smaller outer wing, relative to a similar sized conventional aircraft translates into an increased lift-to-drag ratio, since it is proportional to the wetted aspect ratio; this aspect ratio increases due to its inverse proportionality to the wetted area [29, 31, 59, 53]. The lower wetted area to volume ratio for larger BWBs in comparison to conventional aircraft also adds to the benefit. Interference drag is reduced due to the elimination and reduction of junctions which exist between the wings and fuselage on conventional aircraft [55, 53, 54, 63, 50, 46], resulting in a more streamlined shape for the BWB. The absence of the horizontal tail also implies a reduction in the corresponding friction and induced drag penalties, further increasing the lift-to-drag ratio [1]. The naturally area-ruled shape of the BWB means higher cruise Mach numbers are more easily attainable without changes in the basic configuration geometry [60, 29]. In fact, the BWB's cross-sectional area variation resembles that of the body of minimum wave drag due to volume, the Sears-Haack body, translating into wave drag reductions at transonic speeds [60, 29].

With engines partially embedded in the BWB aft-body, an advantage unique to this

configuration arises: the potential for boundary layer ingestion from a portion of the

centerbody upstream of the engine inlet. Not only does this aft-location of the engines effectively balance the airframe and offset the weight of the payload, furnishings, and systems, but it also ensures that such Boundary Layer Ingestion (BLI) technology has greatest effect since the boundary layer is fully developed towards the rear of the wing [8]. In addition, through the reduction of ram drag, this BLI technology can provide improved propulsive efficiency [31, 29], as well as reductions in required thrust and fuel burn [8]. Finally, the potential for further drag reduction through passive and active laminar flow control via wing shaping and laminar flow technology on the engine nacelle and lifting surfaces is present, as the BWB configuration is well-suited for such technologies. This would imply potentially substantial reductions in skin friction drag [53].

2.6.2 Aero-structures

Since the lift-generating fuselage extends spanwise, the lift and payload are much more in line with each other on the BWB than on a conventional aircraft [50]. Essentially, the passenger cabin is used as a wing bending structure. Consequently, the cantilever span of the thin outer wing is reduced, and the BWB weight is distributed more optimally along the span [31]. This integration of the thick centerbody with the outer wing translates into reduced bending moments and thus reduced structural weight [53, 50, 63]. For the Boeing configuration presented by Liebeck [31], this effect resulted in peak bending moment and shear on the BWB which were half that of a conventional configuration. As mentioned above, this integration also reduces the total wetted area and allows for a long wingspan [52, 31]. As a result, the optimal aspect ratio of the outer wing can be slightly greater than that for conventional wings [1]. Thus, not only does the wing have a higher lift-to-drag ratio, but it is also structurally efficient [52, 31, 1].

2.6.3 Noise Reduction

Even prior to the implementation of specific acoustic treatments, the BWB configuration has a low acoustic signature [29]. For this reason, the BWB was selected for the MIT/Cambridge Silent Aircraft Initiative project (SAI), which had the goal of designing an aircraft with reduced noise [8]. The airframe has no tail, smooth lifting surfaces and minimally exposed edges and cavities, contributing to its low-noise nature. The BWB is more of a noise-shielded configuration than current conventional aircraft on which the engines hang below the wing [8]. In the case where engines are located on the aft-body of the BWB [52, 8, 63], the inlets are hidden from below by the centerbody, which also serves as a shield for forward radiated fan noise. Furthermore, engine exhaust noise is not reflected from the under surface of the wing, benefiting both the passengers and areas surrounding airports [31, 29, 1]. Due to more specific features of the Boeing design, air- frame noise is further reduced through the absence of slotted flaps | due to the low wing loading | for the trailing edge high-lift system and all the mechanisms which support them [31, 29].

2.6.4 Marketing And Manufacturing :

In terms of passenger comfort levels in the BWB, this configuration's vertical cabin walls might present a more spacious environment than the current curved walls of conventional aircraft [29]. Liebeck [29] compares this design and spacious environment to that in a railroad car. Direct operating costs per seat/mile for the BWB are also estimated to be 15% lower than current conventional designs [1]. Due to the simplicity of the BWB configuration, such as the elimination of fillets and joints of highly loaded structures at 90 degrees to each other, a significant reduction in the number of parts | on the order of 30% | has also been estimated [30, 29]. A two-fold sense of commonality is another design constraint considered by the Boeing team [30] as a result of the BWB's unique capability to be stretched and re-configured

2.6.5 Size:

commonality between different sizes of the BWB in order to create a family of aircraft.

2.6.6 Application:

commonality between military and commercial applications.

For the former, the aircraft can be stretched laterally, enabling the addition of span and wing area while increasing the payload. This advantageous capability is not afforded by conventional aircraft which are longitudinally stretched to increase payload [29]. Specifically, commonality between 250-passenger and 450-passenger versions has been studied, with the outer wings and nose/cockpit section being common between members of this aircraft family. The necessary fuel volume in the outer wing is adequate for all members of the family, and the modular centerbodies are aerodynamically smooth and balanced. Furthermore, such commonality offers 23% reduction in non-recurring costs and 12% reduction in recurring costs compared to the stand-alone cases for the 250- and 450-passenger versions. Such cost reduction would likely increase with the inclusion of additional sizes of BWB, such as a 350-passenger version [30, 29]. For the Boeing cabin design, this commonality between families also extends to the interior with the growth concept in place, since the cabin cross-sections would be the same between the different aircraft. For airlines, these benefits mean fleet mix requirements can be easily accommodated, manufacturing learning curve penalties are reduced, and maintenance and life-cycle cost savings increased. All of this is achieved through a natural variation of the span and wing area with weight in order to maintain aerodynamic efficiency | an advantage that is possible with this configuration [29]. With respect to the commonality of applications, aircraft applications have also been demonstrated for a variety of military applications including freighter, stand-o_ bomber, troop transport, and tanker; details of military BWBs can be found in [30].

In addition to these possibilities for commonality, the BWB's previously-noted, naturally area-ruled shape could also reduce manufacturing costs associated with conventional aircraft, which must be manufactured with a varying cross-section, `coke-bottle' fuselage in order to achieve area-ruling [60]. This highlights the potential for the BWB to perform at higher speeds at lower costs. Further cost savings are implied since the interior configuration of a BWB is no longer a challenge. In contrast, a conventional aircraft with a varying cross-section will also have varying seats abreast along the area-ruled portion of the fuselage [30]. In the case of the SAI design, the increased aerodynamic and structural efficiency are features which could help offset potentially higher operating costs of a silent engine design [8].

2.6.7 Safety :

The rear location of engines on the BWB places shrapnel from a failed engine behind the pressure vessel, most flight controls, systems and fuel tanks. The pressure vessel, due to its unique structural requirements and the necessity to handle both wing bending and pressure loads, must be robust and will likely have substantial crashworthiness [31, 29]. In addition, in certain configurations, the passenger compartment and fuel are separated by broad cargo bays [31].

2.6.8 Stability And Flight Control :

Liebeck noted that a complicated high-lift system is not required for the Boeing design due to the low effective wing loading of the configuration. Redundance and reconfigurability of the trailing edge flight controls for this design are also discussed [31]. Furthermore, a reduction in the secondary power required by the control system is also demonstrated [30].

2.6.9 Other :

Other potential benefits include increased loading and o_-loading times due to the shorter fuselage length on a medium-sized (200-passenger) BWB [48], as well as a shorter take-off field length without the need for complicated high-lift devices [47].

2.7 Disadvantages of BWB :

2.7.1 Aerodynamics :

For instance, atypical transonic airfoils of high thickness to chord ratio | up to about 17% in the Boeing designs [29] | are required inboard to accommodate passengers, cargo and landing gear. Furthermore, adding to the difficulty of the design of such airfoils, this thickness to chord ratio must be maintained along a considerable portion of the chord length [52, 29]. This poses problems for maintaining low drag [76]. In addition, due to deck angle limitations, the centerbodyairfoils must be designed to generate the necessary lift at angles of attack which are consistent with deck angle requirements [52, 59, 29]. Supersonic flow on the lower surface of the BWB is another challenge, which is not typical on the conventional configuration [52]. Smooth transition from the thicker center- body airfoils to the thinner outer wing airfoils can also be cause for difficulty, particularly for medium-sized 200 passenger BWBs since the transition could be more abrupt for such smaller aircraft [48]. Additionally, the benefit of the reduced wetted area may not hold in all cases; for instance, Pambagjo et al. pointed out that achieving the wetted area reduction could be more challenging in the case of a medium-sized BWB aircraft, which was, in fact, found to have a higher wetted area when compared to conventional aircraft [48]. Finally, while BLI technologies and embedded engines sound promising, challenges with the integration of the engine and airframe and incorporation of these technologies include the design of low-loss inlet ducts, the control of the inlet flow distortion, and the turbo-machinery integration [8]. In the aerodynamic design of the aircraft, manufacturing constraints must also be factored in: complex, three-dimensional shapes which might be expensive and difficult to manufacture must be avoided with smooth, simply curved surfaces being favoured [59, 29].

2.7.2 Propulsion

Additional difficulties of aft-mounted engines and propulsion and airframe integration exist, since engine integration affects several disciplines more directly than is the case for conventional aircraft [76, 31]. Indeed, interaction between the wing, control surfaces, and engines increase the complexity of the design of this region [59]. Liebeck et al. Explore solutions for this issue in [31].

2.7.3 Structures

A key challenge is posed by the BWB's non-cylindrical pressure vessel, which must be light-weight yet capable of handling both the wing bending loads as well as the cabin pressure loads. As shown in [41], a box-type BWB fuselage could have stress about an order of magnitude higher than the stress in a cylindrical pressurized fuselage. The increased stresses in such a pressure vessel naturally lead to increased structural weight [31, 74]. Mukhopadhyay et al. [42, 41] and Velicki et al. [73] discuss detailed concepts considered specifically for the BWB configuration. Mukhopadhyay et al. study different concepts in order to determine the optimal fuselage configuration for the BWB including multibubble fuselage models: two, three, four, and five bubble models [42, 41]. Through this study an overall weight reduction of 20-30% compared to using all at surfaces could be achieved through the proper integration of partially cylindrical surfaces in pressurized fuselage design. In [41], a Y-braced 480-passenger aircraft fuselage which develops into a modified fuselage in which the Y-brace is replaced by a vaulted shell is also discussed. Velicki et al. present the technology likely incorporated in this Y-braced configuration: Pultruded Rod Stitched Efficient Unitized Structure (PRSEUS) | a technology specifically designed, tailored and optimized for the BWB airframe [73]. Features of this technology include continuous load paths in two directions, accommodating the unique spanwise and streamwise load paths of the BWB fuselage, thin skins which operate in the post-buckled design regime, and stitched interfaces to arrest damage propagation. The pressurized shell elements (skin panels and frames) have also been found to be 28% lighter for the PRSEUS concept than comparable sandwich panel designs.

2.7.4 Stability And Flight Control :

The integrated nature of the BWB, along with the elimination of the tail, means that interactions between inertial forces, aerodynamic loads, elastic deformations and the flight control system responses may have great impact on the performance and stability of the aircraft [29, 74, 66]. Several issues arise: the aircraft must be balanced while ensuring control deflections do not adversely affect the spanload and drag[76]. For larger BWBs, such as those considered by NASA and Boeing, control surface hinge moments are substantial [59]. Thus, if the aircraft is unstable and dependent on active flight controls, secondary power requirements could be prohibitive [76, 59, 29].

2.7.5 Marketing And Manufacturing :

While the BWB might present a more spacious environment, there are some Potentially negative aspects that make marketing of this configuration a challenge. First, with a window only in each main cabin door and no other windows on the cabin walls, passengers might be uncomfortable in a BWB. A proposed solution is to use at display screens connected to an array of digital video cameras to make every seat a window seat [29]. Secondly, given the lateral offset from the center of gravity, the ride quality could deteriorate in the outer portions of the BWB. Boeing has performed a series of tests in which piloted flight simulator tests of the BWB-450 and B747-400 using the same pilots and flight profile were carried out for different cases. The comparisons found the ride quality only decreased slightly | about 4% using the NASA Jacobsen ride quality model to determine passenger satisfaction with the ride | for both the best and worst seats on both aircraft [30, 29]. Finally, a minor marketing issue with respect to commonality in a family of BWB aircraft is extra weight on smaller members of the family compared to stand-alone BWB models [30]; however, a relaxation in the requirement that the members have common part numbers permits a skin gauge change, reducing the weight penalty substantially [29].

2.7.6 Certification :

Finally, certification of the BWB might be hindered due to concerns of efficient emergency egress [29]. This could be more problematic for larger BWBs where the distance from the exits increases [1] and lack of clear views of the different exits on larger BWBs will create challenges for cabin crew redirecting passengers [10]. However, both Bolsunovsky et al. and Liebeck argue that procedures compliant with FAR-25 can be implemented [1, 29]. Liebeck argues that passengers have a direct view of one or more exits, without requiring a 90 degree turn to reach the door from the aisle. This is accommodated by the fact that the Boeing design has a main cabin door directly in front of each aisle and an exit through the aft pressure bulkhead at the rear of each aisle. In addition, four spanwise aisles intersect with these longitudinal aisles [29]. Both computer simulations and full- scale evacuation trials carried out by Galea et al. for a 1000+ passenger BWB aircraft showed that improved visual access and awareness of the aircraft layout are key to efficient egress in emergency situations. Fire simulations found 12 fatalities deemed inevitable but independent of the cabin architecture [10].

2.7.7 Other :

Other issues include landing approach speed and attitude and buffet and stall characteristics [59, 29]. In addition, other studies of BWB have shown engines arranged on pylons under the wing [1], which would eliminate a lot of the benefits outlined with respect to noise and drag reduction previously discussed.

several critical problems unique to the BWB must be addressed to advance the concept beyond a preliminary design phase :

Inboard wing design : the inboard portion of the wing contains the passenger cabin and cargo areas within thick , large chord, transonic airfoils, reaching thickness to chord ratios (t/c) of -18% . cabin height leading edge doors and rear spar impose that the thickness be maintained along a considerable length of the chord . deck angle limits are a consideration . Shock strength is of major concern on the centerbody . supersonic flow on the lower surface is uncharacteristic of conventional wing design and must be investigated . pillowing of the pressurized outer skin results in modified aerodynamic shapes .

Kink region design : the portion of the wing which blends the thick, inboard airfoils and thin, supercritical, outboard wing is referred to as the kink . design problems in this region include surface smoothness , lift carry over from the centerbody , shock strength and sweep with possible separation, and buffet tailoring .

Trim : one of the more critical issues to be addressed on the BWB is cruise trim . this is a multidisciplinary problem influenced by the location of the center of gravity (CG) of the aircraft and the required stability levels . it is desired that the airplane be trimmed in the mid-cruise configuration at nominal CG limits with minimal control deflections . detailed pressure distribution design on the centerbody and outboard airfoils , planform layout , and determination of the optimal span loading are important .

2.8 PREVIOUS WORK

2.8.1 NASA Projects (USA) :

In the hopes of setting in motion `a renaissance for the long-haul transport', a study

of the BWB configuration [31] began with a focus on aerodynamics [52] and evolved to more detailed, multi-disciplinary considerations over the years [30, 29, 31, 59]. A preliminary comparison consisting of a streamlined disk versus a tube and progressing with the addition of key aircraft components showed a potential total wetted area reduction of about 33%, which translates into an increased lift-to-drag ratio and motivated further study of this concept [29]. The initial development and feasibility study involved the set-up of a NASA-industry-university team in 1994. This team conducted a 3-year study demonstrating the commercial and technical feasibility of the BWB concept. Members of the team included Mc-Donnell Douglas as program manager, NASA Langley Research Center, NASA John H. Glenn Research Center at Lewis Field, Stanford University, University of Southern California, University of Florida, and Clark-Atlanta University. Several design constraints were considered in the design of an 800-passenger BWB with a 7000 nautical mile range: volume, cruise deck angle, landing approach speed and altitude, buffet and stall, trim, power for control surface actuation, and manufacturing. For aerodynamics, Navier-Stokes computational fluid dynamics (CFD) methodology in both inverse design and direct solution modes were employed to define the final BWB geometry. In addition, transonic and low-speed wind tunnel tests were carried out at NASA Langley Research Center's National Transonic Facility, resulting in excellent agreement between experimental and computational results. For structures, two concepts were studied: a thin, arched pressure vessel above and below each cabin which takes the load in tension and is independent of the wing skin, and a thick sandwich structure for both the upper and lower wing surfaces which handles both cabin pressure loads and wing bending loads. For the former, a potential pressure leak is a point of concern. In this case, Mukhopadhyay et al. state that the outer ribbed shell provides adequate redundancy and is found to be strong enough to withstand operational cabin pressure [42]. However, Liebeck argues that once sized to carry this outer pressure load, the outer wing skin is sufficient, eliminating the need for an inner pressure vessel; consequently, the thick sandwich concept was chosen for the centerbody structure [29]. More recently, Velicki et al. have proposed the PRSEUS concept described earlier [73]. Boundary layer ingestion studies were carried out at both University of Southern California and Stanford University, with the latter performing multidisciplinary optimization studies of the BWB engine inlet concept based on the wind-tunnel simulations carried out by the former [29, 31]. Using a 6% scale flight control testbed built at Stanford University, low-speed flight mechanics were explored and excellent handling qualities within the normal flight envelope were demonstrated. Overall, with significant weight reduction and one less 60,000lb class engine, the fuel burn per seat mile was found to be 27% lower compared to a conventional aircraft.

This study subsequently developed into the study of the BWB-450 | a 450-passenger aircraft deemed more in line with market forecasts and a reasonable comparison to existing aircraft such as the A380, B747, and A340. Using Boeing's proprietary code, WingMOD, MDO was carried out with a vortex-lattice code and monocoque beam analysis coupled to give static aeroelastic loads. A new class of transonic airfoils was designed.

These airfoils not only smoothened and flattened the geometry for simplified manufacture, but also accommodated the cross-sectional area requirements for payload. Structurally, an 18% reduction in the BWB-450's MTOW relative to an A380-700 was achieved. The fuel burn per seat was 32% lower than the A380-700. Aspects of stability, propulsion, environment and performance are discussed in more detail by Liebeck [29], as are unique opportunities and challenges | as discussed in the previous section: manufacturing part count, family and growth opportunities, speed opportunities, passenger acceptance, ride quality and emergency egress [29].

In addition to these studies, as part of the NASA Revolutionary Aerospace Systems Concepts Program, the Quiet Green Transport (QGT) study [17], aimed at developing and evaluating commercial transport aircraft concepts that significantly reduce or eliminate aircraft noise and emissions, as well as identifying technology advances essential to the feasibility of the concepts, considered the BWB configuration with distributed hydrogen fuel cell propulsion. Assuming the availability of certain advanced technologies, project benefits relative to today's conventional aircraft include complete elimination of all aircraft emissions except H2O, the possibility of eliminating the formation of persistent contrails, 10% reduction in the area exposed to noise levels of 55dBA and greater during takeoff and landing operations, and 8 to 22dB EPNL reduction in noise at FAA noise certification points. Despite these advantages, several areas will need to advance significantly to realize the full potential of the concept. For instance, even with 25-30 year projected improvements, conventional aircraft engines are still lighter than the fuel cell based system which would rely on liquid hydrogen [17]. While the concept might be difficult to achieve at this point, the versatility of the BWB is clearly demonstrated.

2.8.2 Multidisciplinary OptimisationOf A Blended Wing Body (MOB) Project (Europe)

The collaborative Multidisciplinary Optimisation of a Blended Wing Body (MOB) [40]

project between universities, research institutes and companies across the UK, Germany, Netherlands, and Sweden has the primary goal of developing a variety of discipline-based (aerodynamics, structural, aero-elastic and flight mechanics) commercial or proprietary programs and tools to enable a range of studies from preliminary, rapid assessments of initial configurations to high-quality, expensive computational simulations and assessments by distributed design teams. Essentially, this distributed yet integrated system, the Computational Design Engine (CDE), is a multi-level, multi-site, multidisciplinary design tool. A secondary goal of the project is to apply the CDE to the BWB aircraft. At Sheffield University, Qin et al.'s contributions to the MOB project include various aerodynamics studies of the BWB [53]. Both high-fidelity RANS evaluation with a Baldwin-Lomax algebraic turbulence model and Euler equations are used in the design process, with key considerations being wave drag, spanwise load distribution, aerofoil section design and 3D shaping for performance improvement. First, an inverse design of the spanwise loading employed a panel method with three target loadings: elliptical | reduces induced drag, triangular | reduces wave drag, and an average of the two | a compromise, with the goal of alleviating high wave drag by shifting the load inboard.

Significant wave drag reduction on the outer wing was achieved for the new twist distributions with the averaged distribution having the minimum total drag and thus, the highest aerodynamic efficiency. The loading on all three designs is moved inwards towards the centerbody relative to the baseline loading, with the highest centerbody loading for the triangular distribution and lowest for the elliptical. Structural and stability advantages to the averaged distribution are highlighted in [53, 54]. Subsequently, starting from the twist inverse design results, airfoils were mapped from 3D to 2D, optimized, and mapped back to 3D. The resulting geometry had a 20% increase in lift-to-drag ratio compared to the baseline; however, the high sweep and 3-dimensionality of the BWB shape implies 2D optimization cannot fully capture the potential of shape optimization, leading to the _nal portion of the study: 3D Euler aero surface optimization, incorporating a twist and camber optimization with pitching moment constraint and 3D surface optimization with trim constraint. The twist inverse design previously obtained was used as the starting geometry. Minimizing pressure drag was deemed crucial since it dominates the total drag due to the lower surface to volume ratio. In addition, the optimal spanwise lift distribution for best aerodynamic performance should be a _ne balance of induced drag due to lift and wave drag due to shock wave formation at transonic speeds, such as the average elliptical/triangular distribution studied. As such, the elliptic distribution should no longer be the target for minimum drag design, unless wave drag can be eliminated by the design optimization, in which case an elliptic distribution may still be favourable for aerodynamics [63]. Further studies done by Qin et al. involve BWB configurations with forward swept wings [63], varying the outer wing sweep angles, defined as the leading edge sweep of the wing, from -40 degrees (forward sweep) to 55 degrees (backwards sweep), keeping planform fixed and using a pitching moment constraint. The lift-to-drag ratio for forward swept wings was low and relatively constant as forward sweep angles were increased. In this case, the wing sweep cancels with the sweep of the body, creating intense shocks at the junction between the two. Increased contributions to the wave drag result in the trend observed. In contrast, for 0 to 55 degrees, the lift-to-drag ratio increased substantially and peaked at 45 degrees. Other interesting studies include the control of shock waves on the BWB via 3D shock control bumps [78].

2.8.3 Tohoku University (Japan) :

Pambagjo et al.'s research [47, 48] aimed to design a regional BWB aircraft for about 224 passengers, a cruise Mach number of 0.8 and a range of 2000 nautical miles. Using a Navier-Stokes flow solver with a fully turbulent boundary layer (Baldwin-Lomax turbulence model), an inverse design and target pressure specification technique were carried out. The first step of this process was to design the target pressure distribution based on the required aerodynamic performance. For the initial design [48], the specified target pressure distribution was obtained with discrepancies at the leading and trailing edges. The outboard wing was inversely designed with the goal of elliptical loading, while ensuring sufficient fuel volume. The inboard section was designed as thin as possible without sacrificing too much of the internal space and comfort. For the follow-up design [47], shocks were eliminated on the upper and lower surfaces and agreement with the target elliptical span loading was improved; however, the pitching moment was slightly higher for this configuration than for the final configuration initially presented [48].

2.8.4 TsAGI (Russia) :

This work | done in conjunction with Airbus and Boeing | placed emphasis on the rationale behind the main design of flying wing configurations and developing and analyzing alternative configurations, while taking the aerodynamic and structural concepts into consideration. The analysis process consisted of three stages: first, development of a baseline configuration to define the project requirements; second, development of three candidate concepts: an integrated wing body (IWB), a lifting-body configuration, and a pure flying wing | all designed for the same requirements for a fair comparison; and third, detailed computational and experimental studies for the most promising layout. With a design mission for 750 passengers, 13,700 km range and cruise Mach number of 0.85, comparisons of the alternatives to the conventional configuration on the basis of aerodynamics, weight, fuel efficiency, etc., led to the selection of the IWB configuration. This configuration maintains the high L/D advantage of a flying wing at about 24.5 for a Mach number of 0.85, maintains structural efficiency, and meets all the main requirements of FAR-25 [1].

2.8.5 MIT/Cambridge Silent Aircraft Initiative (SAI) (USA/UK) :

Given its aerodynamic advantages, noise reduction, and characteristic cost savings, the BWB configuration was an ideal option for SAI's goal of designing an aircraft which is inaudible outside the airport boundary in a typical urban area [69, 8, 9]. Diedrich et al. combined WingMOD [75], an established MDO tool also utilized by Boeing, which incorporates aerodynamics, loads, performance, weights, balance, stability and control considerations with first principles and empirically-based design and acoustic prediction methods, to explore an unconventional BWB aircraft configuration to achieve the stated goal, while also ensuring competitive performance with next-generation aircraft such as the B787 [8]. Based on a design mission of 215 passengers, a range of 5000 nautical miles, a cruise Mach number of 0.80, and technology levels consistent with 2030 entry to service, an optimized aircraft which achieves significant noise reduction compared to similar, existing commercial aircraft was obtained; however the goal of being inaudible outside of the airport perimeter was not achieved and further technology developments were deemed necessary. The optimized aircraft fuel burn was competitive with B787-type aircraft, as desired. Other interesting outcomes of the project include research into the use of leading-edge carving which enables the entire planform to generate lift (an `all-lifting' design) and produces a final aircraft which is both balanced and statically stable | without the use of a reflexed airfoil, for instance [69, 62]. Additional research into landing gear studies [56], Boundary Layer Ingestion studies [51], continuous descent [58] and surface roughness [32] are examples of the types of projects that have and are developing out of this project.

2.8.6 Universidad Politecnica de Madrid (Spain) :

Martinez-Val et al. [34] showed a medium-sized C-wing flying wing for 300 passengers was operationally efficient and preferable to conventional airplanes of similar capacity. The horizontal stabilizers incorporated on the C-wing portion of the body are considered useful for stability and control improvements. Compared to the A330-200 and B777-200, the take-o_ and landing field lengths are shortened for the C-wing. Fuel burn is found to be 19.8g/passenger-km | less than the 21.5 and 23.5g/passenger-km for the A330 and B777, respectively. Extending this work, a comparison of the C-wing with a U-wing flying wing aircraft (a flying wing with vertical winglets) was carried out with a study of how relevant, emerging technologies such as laminar flow control (LFC), vectored thrust, and active stability can provide additional improvements [34]. The U-wing configuration along with the emerging technologies resulted in a fuel efficiency of 14.6g/passenger- km | the lowest of all the options considered. Overall, both the C-wing and U-wing configuration present significant advantages over the conventional layouts; however, it is important to note that while the U-wing configuration is lighter, for this design, it requires vectored-thrust which is currently unavailable for civil aviation [34].

2.8.7 Airbus Projects (Europe)

Airbus is involved in various projects with ONERA and DLR as major collaborators, in addition to other companies and institutions. The Very Efficient Large Aircraft (VELA) project has Airbus as the leading company with 17 European partners from industry and research working together to extend their knowledge of BWB configurations [67]. AVECA, a national project carried out in close collaboration between Airbus and ONERA, focuses on lower capacity flying wing aircraft since this topic has not been as extensively evaluated as the larger counterparts [38]. Finally, the New Aircraft Concepts Research (NACRE) project is also led by Airbus with 36 partners [39, 38]. As part of VELA, each configuration studied in [67] was defined by two leading edge sweep angles on the inboard and outboard wing sections and a corresponding cabin geometry. Constraints included volume, deck cabin angle, and stability considerations. A DLR finite volume flow solver, FLOWer, which solves the 3D compressible Euler equations in integral form, was employed in this two part process consisting of 2D multipoint airfoil design optimization and a 3D twist and chord-length optimization. The 3D optimization employed GenOpt-software as the optimizer - specifically, the Nelder-Mead Simplex algorithm. Eight design variables, such as the change of twist and chord length with constant thickness to chord for the three outer wing sections, were used. Additional details can be found in [67]. The flow analysis was performed in Euler mode with stripwise boundary layer analysis, using the integral method for friction drag estimation. Sixteen percent drag reduction with constant lift was achieved. The shock wave on the outer wing was modified from a strong single shock close to the trailing edge to a weaker double shock system; consequently, the drag reduction was dominated by wave drag reduction (63%). This wave drag reduction was achieved by increasing chord lengths in the outer wing and adjusting twist angles [67]; however, since wave drag was still present, the optimum lift distribution was not of elliptical form, which might be expected if total minimum drag is achieved [39]. Additional research includes the determination of static and dynamic derivatives for stability and handling characteristics, along with high-speed testing, to create an experimental database for aerodynamic performance assessment and Semi-Buried engines (SEBU), through which engine installation effects are studied [39]. The AVECA project [38] studied lower capacity BWB configurations, aiming to design viable flying wing geometries considering aerodynamic cruise performance, longitudinal trimming constraints and geometric constraints such as cabin and landing gear volumes. Specific details of the design mission were excluded. Navier-Stokes, multiblock, structured flow solves for Mach 0.85, Re/c of 172.2 million per meter (Note: the units of Re/c are not specified in the paper and are inferred from geometry images) were carried out using the ONERA elsA code. Some of the goals included increasing the lift-to-drag ratio at the design point and avoiding strong shocks on both sides of the wing while meeting the various geometric constraints. A manual, iterative strategy with a total of 60 configurations was used to design the inboard section and ultimately suppress the shock. For the outboard section, optimization driven by the gradient algorithm, CONMIN, was used to define the twist distribution of the configuration.

The total drag, considering constraints on lift and on the location of the center of pressure, made up the objective function. Two design points were considered: Mach 0.85, Re/c of 172.2 million per meter and Mach 0.87, Re/c of 176.3 million per meter. For the first point, a 9.5% and 39.1% decrease in viscous and wave drag was achieved, respectively; while for the second point, these reductions were 18.9% and 37%, respectively [38]. Additional research under the NACRE project includes projects such as winglet design for large capacity flying wing configurations [38]. The winglet, equipped with control surfaces, is expected to provide substantial drag reduction and additional lateral stability.

This concept with plain flap control surfaces is studied on a VELA2 configuration to demonstrate the efficiency of a winglet and to model its effects on control derivatives.

2.9 LIST OF BLENDED WING BODY AIRCRAFTS

2.9.1 Westland Dreadnought

The Westland Dreadnought was an experimental single-engined fixed-wing monoplane design for a mail plane created to trial the aerodynamic wing and fuselage design ideas of Woyevodsky. It was designed and built by British aircraft manufacturer Westland Aircraft for the Air Ministry. Only a single aircraft was ever built,[1] and it crashed on its initial flight, badly injuring the test pilot.

Design and development

The Dreadnought was distinct for its futuristic design and method of construction, based on the theories of the Russian inventor N. Woyevodsky. After preliminary tests of the idea were tried in a wind tunnel and met with some degree of success, the design was given to Westland Aircraft to construct an aircraft. The design at the time was for a 70 ft wingspan twin-engine aircraft.

The design was aerodynamically advanced, featuring a continuous section over all parts of the aircraft, including the fuselage and, unusually for British aircraft at that time, had no form of wing bracing.

Construction was all-metal, comprising drawn channeling with a skin of corrugated sheet panels. The method may be compared to the modern stressed skin construction.

Another advanced feature was the fail-safe ejection system.

Although conceived as a twin-engined type with retractable undercarriage, the design that emerged was fitted with a single 450 horsepower Napier Lion II 12 cylinder engine that allowed the Dreadnought speeds of up to 102 miles per hour.[1] and fixed undercarriage.[2]

Operational history

On completion of the Dreadnought, pilot Arthur Stewart Keep carried out taxi trials and short airborne hops. On 9 May 1924,[3] he took off for its first flight test. While the aircraft was initially stable, it soon became clear that Keep was losing control, and not long after, at a height of approximately one hundred feet, the Dreadnought stalled and crashed. Thrown from the aircraft,[4] Keep sustained severe injuries, and later had both legs amputated.[5] He remained with the company and did not retire until 1935.[6] After this failure, the Dreadnought design was abandoned, although the ideas that were conceived and used in its making were visibly an advancement in aircraft and are appreciated as such in the present day.

2.9.2 Stout Batwing

Batwing was a name given to at least two aircraft developed by William Bushnell Stout.[1]The first was an experimental low aspect ratio flying wing. The aircraft used wood veneer construction and was an early example of cantilever wing design. The internally braced wing was also one of the first American aircraft designed without drag-producing struts.The second was the Batwing Limousine, a three-seat cabin monoplane with a conventional fuselage and high-mounted wing.

Development

During World War I, William Bushnell Stout was employed by Packard in 1917 when he was appointed as a technical advisor to the War production board who in turn gave Stout a contract to develop an aircraft. Funded by the Motor Products Corporation, Stout developed the "Batwing" aircraft hoping to sell the aircraft to the United States Army Air Service.[2] Stout first experimented with an all-wood flying wing glider, the "Batwing Glider", tested at Ford Airport in 1926.[3] Stout's design was nicknamed "Bushnell's Turtle" (a reference to the unrelated David Bushnell's *American Turtle* shape).[4]

Design

The Batwing was designed with an unusually broad chord, thick section cantilevered wing with the horizontal stabilizers set very close to the rear of the aircraft.

The wings were covered with a 3 ply wood veneer only 1/20th of an inch thick. The internal bracing consisted of hundreds of spruce struts. Nine spars tested to 1 ton of load each.[5] Like encountering a Junkers F.13, Bill Stout abandoned wood construction for metal corrugated skinning over a metal frame.[6]

To reduce drag, the aircraft employed a cantilever wing without support wires or struts. This required a "thick" wing to build a spar deep enough to support the aircraft. To maintain the thin airfoil sections commonly used at the time, the chord also had to be longer as the wing became thicker. In the case of the Batwing, the chord was almost the entire length of the aircraft. Since the spar did not need to be as thick toward the tips to support the load, the chord decreased further out along the wing, forming an oval shaped wing. As ideal as this was, it caused significant engineering challenge.[7] Further aerodynamic drag reductions came from having the water-cooled engine embedded into the wing with retractable radiators.[8]

The pilot sat in an open cockpit placed at the top of the aircraft. Visibility was restricted downward by the wing. The Batwing was the first example of a cantilevered wing and veneer skin designed and built in the United States.[9]

Operational history

The mockup of his first thick winged aircraft design was built at the Widman woodworking plant in Detroit, Michigan. The 150 hp engine was acquired from Charles Warren Nashwho had an interest in the project.[10] The first flight was in Dayton, Ohio in 1918. The pump shaft on the engine was broken, but the plane was flown anyway. Although the flight was successful, the test pilot Jimmie Johnson commented that the aircraft was too dangerous to fly because of the poor visibility. Stout later called the visibility "abominable". The test aircraft was put into storage. Soon afterward, Stout submitted British patent #149,708, with a Batwing aircraft with the corners squared off rather than the oval design of the prototype. The updated aircraft was never produced. Stout went on to focus on more conventional aircraft featuring the advancement of all-metal construction, but maintained that the airplane of the future would look like the batwing.

2.9.3 Northrop Grumman X-47A Pegasus

The **Northrop Grumman X-47** is a demonstration Unmanned Combat Aerial Vehicle. The X-47 began as part of DARPA's J-UCAS program, and is now part of the United States Navy's UCAS-D program to create a carrier-based unmanned aircraft. Unlike the Boeing X-45, initial Pegasus development was company-funded. The original vehicle carries the designation **X-47A Pegasus**, while the follow-on naval version is designated X-47B.

Design and development

The US Navy did not commit to practical UCAV efforts until mid-2000, when the service awarded contracts of US$2 million each to Boeing and Northrop Grumman for a 15-month concept-exploration program.[1]

Design considerations for a naval UCAV included dealing with the corrosive salt-water environment, deck handling for launch and recovery, integration with command and control systems, and operation in a carrier's high electromagnetic interference environment. The Navy was also interested in using their UCAVs for reconnaissance missions, penetrating protected airspace to identify targets for the attack waves.

The Navy went on to give Northrop Grumman a contract for a naval UCAV demonstrator with the designation of "X-47A Pegasus", in early 2001. The proof-of-concept X-47A vehicle was built under contract by Burt Rutan's Scaled Composites at the Mojave Spaceport. The Pegasus demonstrator looks like a simple black arrowhead with no vertical tailplane. It has a leading edge sweep of 55 degrees and a trailing edge sweep of 35 degrees. The demonstrator has retractable tricycle landing gear, with a one-wheel nose gear and dual-wheel main gear, and has six control surfaces, including two elevons and four "inlaids". The inlaids are small flap structures mounted on the top and bottom of the wing forward of the wingtips.

The X-47A is powered by a single Pratt & Whitney Canada JT15D-5C small high-bypass turbofan engine with 3,190 lbf (14.2 kN) thrust. This engine is currently in use with operational aircraft such as the Aermacchi S-211 trainer. The engine is mounted on the demonstrator's back, with the inlet on top behind the nose. The inlet duct has a serpentine diffuser to prevent radar reflections off the engine fan. However, to keep costs low, the engine exhaust is a simple cylindrical tailpipe, with no provisions for reducing radar or infrared signature.

The X-47A's airframe is built of composite materials, with construction subcontracted out to Burt Rutan's Scaled Composites company, which had the expertise and tooling to do the job inexpensively. The airframe consists of four main assemblies, split down the middle with two assemblies on top and two on bottom.

The X-47A was rolled out on 30 July 2001 and performed its first flight on 23 February 2003 at the US Naval Air Warfare Center at China Lake, California. The flight test program did not involve weapons delivery, but Pegasus does have two weapons bays, one on each side of the engine, that may be each loaded with a single 500 pound (225 kg) dummy bomb to simulate operational flight loads. The Pegasus was also used to evaluate technologies for carrier deck landings, though the demonstrator did not have an arrestor hook. Other issues related to carrier operations involve adding deck tie-downs without compromising stealth characteristics, and designing access panels so that they would not be blown around or damaged by strong winds blowing across the carrier deck. The J-UCAS program was terminated in February 2006 following the US military's Quadrennial Defense Review. The US Air Force and US Navy proceeded with their own UAV programs. The Navy selected Northrop Grumman's X-47B as its Unmanned Combat Air System demonstrator (UCAS-D) program.[2]

2.9.4 Miles M.30

The Miles M.30 X-Minor was an experimental aircraft, designed by Miles Aircraft to evaluate the characteristics of blendedfuselage and wing intersections

Design and development

Begun in 1939, the design was a scaled-down version of the gigantic Miles M.26 airliner (Miles X) then being developed. The proposed Miles X Airliner was to have had a blended fuselage, eight engines driving four sets of contra-rotating propellers, seating 55 with a range of 3,450 miles (5,550 km). The Miles X Airliner was offered as candidate to the post Second World War Brabazon Report Type 1

Requirement for a trans-Atlantic use but was rejected because the Miles design had only half the seating required.

The small size of the X Minor made it impossible to scale the larger design exactly; the engines were too large and resulted in an aircraft similar in layout but differing inaerodynamics. The X Minor first flew in February 1942, providing Miles with useful data for several years. A larger scale prototype of the X transport was planned but never built.

2.9.5 McDonnell XP-67

The McDonnell XP-67 "Bat" or "Moonbat"[N 1] was a prototype for a twin-engine, long-range, single-seat interceptor aircraftor the United States Army Air Forces. Although the design was conceptually advanced, it was beset by numerous problems and never approached its anticipated level of performance. The project was cancelled after the sole completed prototype was destroyed by an engine fire.

Design and development

Final design

McDonnell engineers returned on 30 June 1941 with the **Model II**, which was also rejected, so it was reworked into the **Model IIa**, which emerged on 24 April 1942. The new design was powered by a more traditional layout, a pair of engines in wing-mounted nacelles with four-bladed propellers in a tractor configuration. However, the design was still quite ambitious; the design team tried to maintain a true airfoil section through the center fuselage, merge the rear portions of the engine nacelles with the wing, and radically fillet all edges of the fuselage and nacelles into the wings in an effort to reduce drag. The design used laminar airfoil sections throughout. McDonnell designers promised that the design would deliver a top speed of 472 mph (760 km/h) with a gross weight of 18,600 lb (8,440 kg), although the anticipated gross weight was soon increased to a somewhat more realistic 20,000 lb (9,070 kg).

On 30 September 1941, the USAAF[N 2] granted McDonnell a $1,508,596 contract, plus an $86,315 fee, for two prototypes, a wind tunnel model, and associated engineering data, The Model IIa was designated as the **XP-67**.[3] The production aircraft was intended to have a pressurized cockpit, a novel innovation at the time. A number of armament configurations were considered including six .50 in (12.7 mm) machine guns, four 20 mm (.79 in) cannon, and even a 75 mm (2.95 in) cannon before the configuration of six 37 mm (1.46 in) M4 cannon was chosen. Power would be provided by two Continental XIV-1430-1 inverted V-12 engines, fitted with turbosuperchargers, and the engine exhaust gasses would augment thrust.

Operational history

On 23 March 1944, flight trials restarted. U.S. Army Air Forces pilots finally got to fly the aircraft on 11 May 1944, and judged the cockpit layout fair and ground handling satisfactory, but deemed the aircraft underpowered due to its poor initial rate of climb, slow acceleration, and long takeoff roll, particularly when operating with only one engine.[6]Other flight characteristics were generally good during gentle maneuvers; stick forces were light, roll rate was adequate, and control was effective at all speeds with good longitudinal stability. However, a tendency to *dutch roll* was prevalent.[5] The prototype also displayed several disturbing behaviors as its stall speed was approached. It began to buffet well above the actual stall speed, it felt tail-heavy in fast turns, and its nose would tuck upwards during the stall. The problems were serious enough that test pilots declined to test the XP-67's spin characteristics, fearing that a spin might be unrecoverable. This irregular and unstable stall behavior has been attributed to advanced aerodynamic principles that were not fully counteracted until the advent of electronic stability controls years later.[6] Although the final flight test report was generally positive, the aircraft's maneuverability was deemed inferior to existing types such as the North American P-51 Mustang.[6]

Upon return to the factory, the cooling ducts were reworked. Several problems were cured during the ensuing test flights, but the engines continued to be plagued by chronic overheating and deficient power output. The XP-67 only reached a confirmed top speed of 405 mph (652 km/h), which was far short of its promised top speed of 472 mph (760 km/h), and was unremarkable compared to other fighters in service at the time.

Project cancellation

Figure 2.10 Head-on view of the XP-67.

On 6 September 1944, the starboard engine of the XP-67 caught fire during a test flight, and test pilot E.E. Elliot executed an emergency landing at Lambert Field in St. Louis, Missouri. He attempted to park the craft pointing into the wind to blow the flames away from the airframe, but the starboard main landing gear brakes failed, pivoting the XP-67 so the flames blew directly towards the aft fuselage. Elliot escaped safely, but the blaze gutted the fuselage, engine, nacelle and starboard wing; the aircraft was a total loss.[7]

The destruction of the lone flying prototype dealt a serious blow to the entire program because the second prototype was only 15% complete at the time. Army leaders decided to reevaluate the XP-67, ultimately deciding on 13 September that it offered no significant advantages over existing fighters already in service.[7] The project was canceled, the remains of the first prototype were scrapped, and work was halted on the second prototype.[N 3]

2.9.6 Lockheed Martin RQ-170 Sentinel

The **Lockheed Martin RQ-170 Sentinel** is an unmanned aerial vehicle (UAV) developed by Lockheed Martin and operated by the United States Air Force (USAF) for the Central Intelligence Agency (CIA). While the USAF has released few details on the UAV's design or capabilities, defense analysts believe that it is a stealth aircraft fitted with aerial reconnaissanceequipment.

RQ-170s have been reported to have operated in Afghanistan as part of Operation Enduring Freedom. It has been confirmed that the UAVs have operated over Pakistan and Iran; operations over Pakistan included sorties that collected intelligence before and during the operation which led to the death of Osama bin Laden in May 2011.[2]

In December 2011, the Iranian armed forces claimed to have captured an RQ-170 flying over Iran. The U.S. military has acknowledged losing an RQ-170 in the region. United States administration asked Iran to return the UAV. Iran denied the request and lodged a complaint to the UN Security Council over

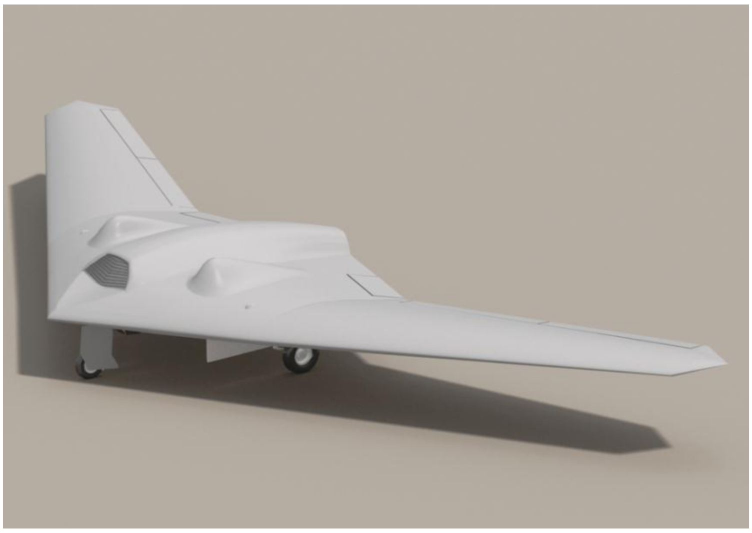

Figure 2.11 airspace violation by the U.S.[3][4][5]

Development

The RQ-170 Sentinel was developed by Lockheed Martin's Skunk Works as a stealth Unmanned Aerial Vehicle (UAV). Journalists have noted design similarities between the RQ-170 and previous stealth and UAV programs such as the RQ-3 DarkStar and Polecat.[6][7] It is a tailless flying wing aircraft with pods, presumably for sensors or SATCOMs, built into the upper surface of each wing. Few details of the UAV's characteristics have been released, but estimates of its wingspan range from approximately 65 feet (20 m)[8] to 90 feet (27 m).[9] In a December 2012 report, journalist David Axe stated that "20 or so" RQ-170s had been built.[1]

Design

The RQ-170 is a flying wing design containing a single (as yet classified) engine and is estimated by *Aviation Week* as having a wingspan of approximately 66 feet (20 m).[15] Its takeoff weight is estimated as being greater than the RQ-3 DarkStar's, which was 8,500 pounds (3,900 kg). The design lacks several elements common to stealth engineering such as zig-zag edged landing gear doors and sharp leading edges, and the exhaust is not shielded by the wing.[15] *Aviation Week* postulates that these elements suggest the designers have avoided 'highly sensitive technologies' due to the near certainty of eventual operational loss inherent with a single engine design and a desire to avoid the risk of compromising leading edge technology.[15] The publication also suggests that the medium-grey color implies a mid-altitude ceiling, unlikely to exceed 50,000 feet (15,000 m) since a higher ceiling would normally be painted darker for best concealment.[15] The postulated weight and ceiling parameters suggests the possible use of a General Electric TF34 engine, or a variant, in the airframe.[15]

On the basis of the few publicly available photographs of the RQ-170, aviation expert Bill Sweetman has assessed that the UAV is equipped with an electro-optical/infrared sensor and possibly an active electronically scanned array (AESA) radar mounted in its belly fairing. He has also speculated that the two undercarriage fairings over the UAV's wings may house datalinks and that the belly fairing could be designed for modular payloads, allowing the UAV to be used for strike missions and/or electronic warfare.[16] The *New York Times* has reported that the RQ-170 is "almost certainly" equipped with communications intercept equipment as well as highly sensitive sensors capable of detecting very small amounts of radioactive isotopes and chemicals which may indicate the existence of nuclear weapons facilities.[17]

Following Iranian claims of downing an RQ-170 near the Afghan border in December 2011, Iranian TV showed video footage of what appears to be an advanced unmanned U.S. aircraft that most closely resembles the RQ-170 UAV. In the footage, a member of the Iranian revolutionary guard released dimensions of the aircraft, including a wingspan of about 26 metres (85 ft), a height of 1.84 metres (6.0 ft), and a length of 4.5 metres (15 ft).[18]

Operational history

The 30th Reconnaissance Squadron operates RQ-170 Sentinels. This squadron, which is based at Tonopah Test Range Airport in Nevada, was activated on 1 September 2005. RQ-170 Sentinels have been deployed to Afghanistan, where one was sighted at Kandahar International Airport in late 2007.[8] This sighting, and the Sentinel's secret status at the time, led Bill Sweetman to dub it the "Beast of Kandahar".[19] The UAV being deployed to Afghanistan, despite the Taliban having no radar, led to speculation that the aircraft was used to spy on Pakistan or Iran: "Phil Finnegan, a UAV analyst at the Teal Group, an aerospace consulting firm, suggests the stealth capabilities are being used to fly in nearby countries. Neighboring Iran has an air force and air defense system that would require stealth technology to penetrate."[20][21]

2.9.7 Lockheed Martin RQ-3 DarkStar

The **RQ-3 DarkStar** (known as **Tier III-** or "Tier three minus" during development) is an unmanned aerial vehicle (UAV). Its first flight was on March 29, 1996. The Department of Defense terminated DarkStar in January 1999, after determining the UAV was not aerodynamically stable and was not meeting cost and performance objectives.[1]

Figure 2.12

Design and development

The RQ-3 DarkStar was designed as a "high-altitude endurance UAV", and incorporated stealth aircraft technology to make it difficult to detect, which allowed it to operate within heavily defended airspace, unlike the Northrop Grumman RQ-4 Global Hawk, which is unable to operate except under conditions of air supremacy. The DarkStar was fully autonomous: it could take off, fly to its target, operate its sensors, transmit information, return and land without human intervention. Human operators, however, could change the DarkStar's flight plan and sensor orientation through radio or satellite relay. The RQ-3 carried either an optical sensor or radar, and could send digital information to a satellite while still in flight. It used a single airbreathing jet engine of unknown type for propulsion. One source claims it's Williams-Rolls-Royce FJ44-1A turbofan engine.[2]

2.9.8 DassaultnEUROn

The DassaultnEUROn is an experimental unmanned combat aerial vehicle (UCAV) being developed with international cooperation, led by the French company Dassault Aviation. Countries involved in this project include France, Greece, Italy, Spain, Sweden and Switzerland. The design goal is to create a stealthy, autonomous UAV that can function in medium- to high-threat combat zones. Comparable projects include the British BAE Systems Taranis, American Boeing X-45 andNorthrop Grumman X-47B, the Indian DRDO AURA, and the Russian MiGSkat.

Figure 2.13

Description

This flying wing stealth UCAV project is the final phase of the French Dassault LOGIDUC 3-step stealth "combat drone" programme. Until June 2005 it had the form of the original Dassault developed Grand Duc vehicle: supersonic two-engined long-range unmanned bomber, capable of performing attacks with nuclear weapons.

2.9.9 Boeing X-45

The Boeing X-45 unmanned combat air vehicle is a concept demonstrator for a next generation of completely autonomousmilitary aircraft, developed by Boeing's Phantom Works. Manufactured by Boeing Integrated Defense Systems, the X-45 was a part of DARPA's J-UCAS project.

Figure 2.14

Development

Boeing developed the X-45 from research gathered during the development of the Bird of Prey. The X-45 features an extremely low-profile dorsal intake placed near the leading edge of the aircraft. The center fuselage is blended into a swept lambda wing, with a small exhaust outlet. It has no vertical control surfaces — split ailerons near each wingtip function as asymmetric air brakes, providing rudder control, much as in Northrop's flying wings.

Variants

X-45A

Boeing built two of the model X-45A; both were scaled-down proof-of-concept aircraft. The first was completed by Boeing's Phantom Works in September 2000.[1] The goal of the X-45A technology demonstrator program was to develop the technologies needed to "conduct suppression of enemy air defense missions with unmanned combat air vehicles."[1] The first generation of unmanned combat air vehicles are primarily planned for air-to-ground roles with defensive air-to-air capabilities coupled with significant remote piloting.

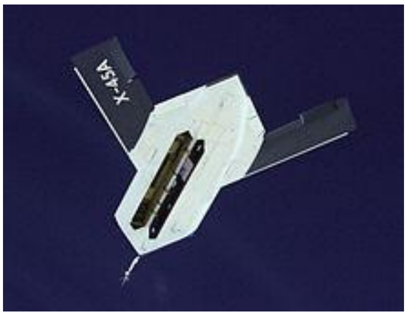

Figure 2.15 X-45A underside with weapons bay door open

The X-45A had its first flight on May 22, 2002, and the second vehicle followed in November of that year. On April 18, 2004, the X-45A's first bombing run test at Edwards Air Force Base was successful; it hit a ground target with a 250-pound inert precision-guided munition. On August 1, 2004, for the first time, two X-45As were controlled in flight simultaneously by one ground-based pilot.

On February 4, 2005, on their 50th flight, the two X-45As took off into a patrol pattern and were then alerted to the presence of a target. The X-45As then autonomously determined which vehicle held the optimum position, weapons (notional), and fuel load to properly attack the target. After making that decision, one of the X-45As changed course and the ground-based pilot authorized the consent to attack the simulated antiaircraft emplacement. Following a successful strike, another simulated threat, this time disguised, emerged and was subsequently destroyed by the second X-45A.[2] This demonstrated the ability of these vehicles to work autonomously as a team and manage their resources, as well as to engage previously-undetected targets, which is significantly harder than following a predetermined attack path.

After the completion of the flight test program, both X-45As were sent to museums, one to the National Air and Space Museum, and the other to the National Museum of the United States Air Force at Wright-Patterson Air Force Base, where it was inducted on November 13, 2006.[1][3]

X-45B/C

Figure 2.16 The newer, larger X-45C

Figure 2.17 X-45C from the side

The larger X-45B design was modified to have even more fuel capacity and three times greater combat range, becoming the X-45C. Each wing's leading edge spans from the nose to the wingtip, giving the aircraft more wing area, and a planform very similar to the B-2 Spirits'. The first of the three planned X-45C aircraft was originally scheduled to be completed in 2006, with capability demonstrations scheduled for early 2007. By 2010, Boeing hoped to complete an autonomous aerial refueling of the X-45C by a KC-135 Stratotanker. Boeing has displayed a mock-up of the X-45C on static displays at many airshows.

The X-45C portion of the program received $767 million from DARPA in October 2004, to construct and test three aircraft, along with several supplemental goals. The X-45C included an F404 engine.[4] In July 2005, DARPA awarded an additional $175 million to continue the program, as well as implement autonomous Aerial refueling technology.[5]

On March 2, 2006, the US Air Force decided not to continue with the X-45 project. However, Boeing submitted a proposal to the Navy for a carrier based demonstrator version of the X-45, designated the X-45N.

X-45N

The X-45N was Boeing's proposal to the Navy's Unmanned Combat Air Systems demonstration project. When it became known that the US Air Force would end funding to the Joint Unmanned Combat Air System program[6] (which included the X-45 and X-47), the US Navy started its own UCAS program.[7] Requirements were defined over the summer of 2006, and proposals were submitted in April 2007.[8]

The first flight of the X-45N was planned for November 2008, had Boeing won the contract.[9] The contract was eventually awarded to Northrop Grumman's proposed naval X-47, thus ending the X-45 program.[10]

The software Boeing developed to allow the X-45N to land and takeoff autonomously on aircraft carriers has recently been installed on the first F/A-18F, which has used it to perform autonomous approaches. All autonomous approaches ended with a wave-off by design. This Super Hornet is expected to be able to hook the carrier's arrester cables autonomously by the 2009 timeframe,[11] setting the stage for carrier-borne UAV operations.

2.9.10 Boeing X-48

The Boeing X-48 is an experimental unmanned aerial vehicle (UAV) for investigation into the characteristics of blended wing body (BWB) aircraft, a type of flying wing. Boeing designed the X-48 and two examples were built by Cranfield Aerospace in the UK. Boeing began flight testing the X-48B version for NASA in 2007. The X-48B was later modified into the X-48C version. It was flight tested from August 2012 to April 2013. Boeing and NASA plan to develop a larger BWB demonstrator.

Figure 2.18

Design and development

Background

Boeing had in the past studied a blended wing body design, but found that passengers did not like the theater-like configuration of the mock-up; the design was dropped for passenger airliners, but retained for military aircraft such as aerial refueling tankers.[N 1][1]

McDonnell Douglas developed the blended wing concept in the late 1990s,[2] and Boeing presented it during an annual Joint AIAA/ASME/SAE/ASEA Propulsion Conference in 2004.[3] The McDonnell Douglas engineers were confident that their design had several advantages, but their concept, code named "Project Redwood" found little favor at Boeing after their 1997 merger.[4][5] The most difficult problem they solved was that of ensuring passengers a safe and fast escape in case of an accident, since emergency door locations were completely different from those in a conventional aircraft.[6]

The blended wing body (BWB) concept offers advantages in structural, aerodynamic and operating efficiencies over today's more conventional fuselage-and-wing designs. These features translate into greater range, fuel economy, reliability and life cycle savings, as well as lower manufacturing costs. They also allow for a wide variety of potential military and commercial applications.[7][8]

X-48

Figure 2.19 X-48B on static display at the 2006 Edwards Airshow

Boeing Phantom Works developed the blended wing body (BWB) aircraft concept in cooperation with the NASA Langley Research Center. In an initial effort to study the flight characteristics of the BWB design, a remote-controlled propeller-driven blended wing body model with a 17 ft (5.2 m) wingspan was successfully flown in 1997. The next step was to fly the 35 ft (10.7 m) wide X-48A in 2004, but that program was later canceled.[9]

Research at Phantom Works then focused on a new model, designated X-48B, two examples were built by United Kingdom-based Cranfield Aerospace. Norman Princen, Boeing's chief engineer for the project, stated in 2006: "Earlier wind-tunnel testing and the upcoming flight testing are focused on learning more about the BWB's low-speed flight-control characteristics, especially during takeoffs and landings. Knowing how accurately our models predict these characteristics is an important step in the further development of this concept."[10]

The X-48B has a 21-foot (6.4 m) wingspan, weighs 392-pound (178 kg), and is built from composite materials. It is powered by three smallturbojet engines and is expected to fly at up to 120 kn (220 km/h) and reach an altitude of 10,000 feet (3,000 m).[10][11] The X-48B is an 8.5% scaled version of a conceptual 240-foot wide design.[11][12] Though passenger versions of the X-48B have been proposed, the design has a higher probability of first being used for a military transport.[12]

Figure 2.20 The X-48C featuring two engines and inboard vertical stabilizers

Wind tunnel testing on a 12 ft wide blended wing body model was completed in September 2005.[13][14] During April and May 2006, NASA performed wind tunnel tests on X-48B Ship 1 at a facility shared by Langley and Old Dominion University.[14][15] After the wind tunnel testing, the vehicle was shipped to NASA's Dryden Flight Research Center at Edwards Air Force Base to serve as a backup to X-48B Ship 2 for flight testing.[16] X-48B Ship 2 then conducted ground tests and taxi testing in preparation for flight.[17] In November 2006, ground testing began at Dryden, to validate the aircraft's systems integrity, telemetry and communications links, flight-control software and taxi and takeoff characteristics.

The second X-48B was modified into the X-48C starting in 2010 for further flight tests.[18] The X-48C has its vertical stabilizers moved inboard on either side of the engines, and its fuselage extended aft, both in an attempt to reduce the aircraft's noise profile; it was to be powered by two JetCat turbines, each producing 80 pounds-force (0.36 kN) of thrust.[19][20] The X-48C was instead modified to use two Advanced Micro Turbo (AMT) turbojet engines in 2012.[21]

Following flight testing of the X-48C in April 2013, Boeing and NASA announced plans to develop a larger BWB demonstrator capable of transonic flight.[22]

Chapter 3 : calculation

3.1 Requirement

1- Range=5500 nmi
2- endurance=2 hours (minimum)
3- stalling speed=76 m/s =250 ft/s
4- cruise speed = 252m/s=826.77 ft/s (mach 0.853 at altitude of 35000 ft)
5- max speed=265.86 m/s = 872.24 ft/s (mach 0.9 at altitude of 35000 ft)
6- landing distance = 2000 m = 6561 ft
7- takeoff distance = 2500 m =8202 ft
8- maximum speed at mid cruise 950 ft/s
9- Rate of climb at sea level 3500 ft/min
10- ceiling 43000 ft

the requirements was chosen based on historical data from aircrafts which has a similar weight like (boeing 777 and A380) of 500 passenger capacity, and this airplane has better performance . so a higher range with good endurance has been selected, and it flies on higher speed at cruise and have lower stalling speed more than aircrafts of historical data with good service ceiling , and it needs less distance to takeoff and to land.

3.2 The weight of an airplane and its first estimate

From requirements :

Range= $5500\ nmi$

$V_{cruise} = 826.77\ ft/s$

$$W_0 = \frac{W_{crew}+W_{payload}}{1-\frac{W_f}{W_0}-\frac{W_e}{W_0}} \quad ...[3.1]$$

3.2.1 Estimation of value $\frac{W_e}{W_0}$

From historical data

Table 3.1 empty weight estimation

Aircraft	W0	We	We/W0
Boeing 707	557000	277000	0.497307
Boeing 717	545000	297300	0.545505
Boeing 727	315000	176650	0.560794
Boeing 737	255000	127520	0.500078
Boeing 747	735000	358000	0.487075
Boeing 757	111000	62000	0.558559
Boeing 767	170000	80602	0.474129
Boeing 777	110000	67500	0.613636
Boeing 787	247000	109700	0.44413

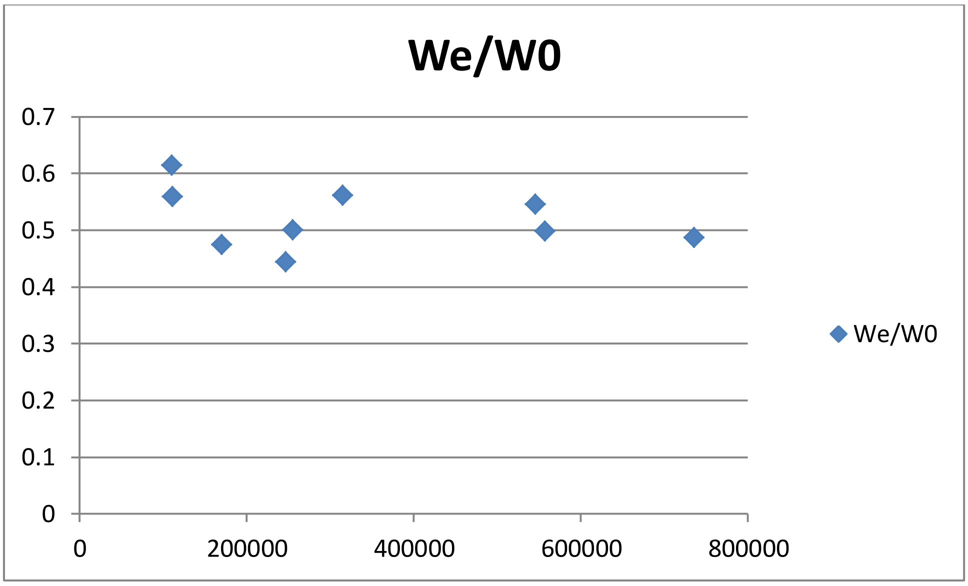

Figure 3.21 empty weight fraction

From that $\frac{W_e}{W_0} = 0.5$ ...[3.2]

3.2.2 Estimation of value $\frac{W_f}{W_0}$

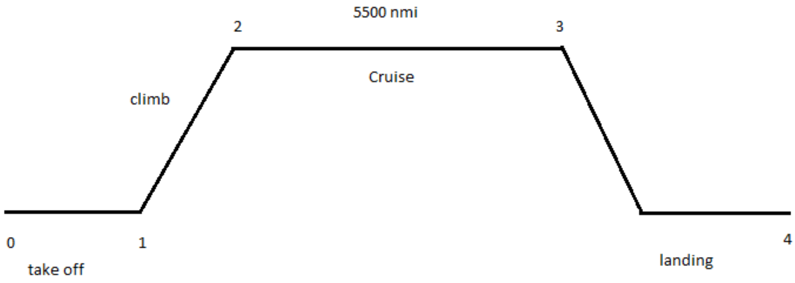

Figure 3.22 mission profile

Cruise-segment mission weight fractions can found by using the brogue range equation:

R=$\frac{V}{C}.\frac{L}{D}.ln\frac{W_{i-1}}{W_i}$ ...[3.3]

$$\frac{W_i}{W_{i-1}} = e^{\frac{-R.C}{V.\frac{L}{D}}}$$

From similar data:

$\frac{L}{D} = 21$ ...[3.4]

$\frac{W_1}{W_0} = 0.97$ ...[3.5]

$\frac{W_2}{W_1} = 0.985$ ...[3.6]

$\frac{W_4}{W_3} = 0.995$ ...[3.7]

At cruise:

R=$5500 \times 6076 = 33418000\ ft$

For high bypass turbofan the specific fuel consumption (C)=0.5 1/hr

$C = \frac{0.5}{60 \times 60} = 0.000139 \ \frac{1}{s}$ ...[3.8]

The most efficient cruise velocity for a jet aircraft occurs at a slightly higher velocity yielding an L\D 86.6% of the maximum L\D

$$\frac{L}{D} = 0.866 \times 21$$

so:

$$\frac{W_3}{W_2} = e^{\frac{-33418000 \times 0.000139}{826.77 \times 21 \times 0.866}}$$

$\frac{W_3}{W_2} = 0.734$ ...[3.9]

Then

$\frac{W_7}{W_0} = \frac{W_1}{W_0} \times \frac{W_2}{W_1} \times \frac{W_3}{W_2} \times \frac{W_4}{W_3}$ ...[3.10]

$$\frac{W_7}{W_0} = 0.97 \times 0.985 \times 0.734 \times 0.995$$

$\frac{W_7}{W_0} = 0.698$ ...[3.11]

$\frac{W_f}{W_0} = 1.06\ (1\text{-}0.698)$

$\frac{W_f}{W_0} = 0.32$ ...[3.12]

a 6% allowance for reserve and trapped fuel

3.2.3 Calculation of W_0

NO of Passenger =500

NO of crew =17

From raymer suggests we choose :

$$W_{crew\ or\ passenger} = 180\ lb$$

$$W_{P.baggage} = 50\ lb$$

$$W_{C.baggage} = 30\ lb$$

$$W_{payload} = 500 \times 180 + 500 \times 50 + 17 \times 30 = 115510\ lb$$

$W_{crew} = 17 \times 180 = 3060\ lb$

The total weight :

$$W_0 = \frac{3060 + 115510}{1 - 0.32 - 0.5} = 658722.2\ lb$$

Fuel weight:

$$W_f = \frac{W_f}{W_0} W_0 = 0.32 \times 658722.2$$

$W_f = 210791.1\ lb$ ...[3.13]

Tank capacity $= \frac{210791.1}{5.64}$

Tank capacity $= 37374.3 gal$ ...[3.14]

$fuel\ tank\ volume = 0.134 \times 37374.3 = 5008.16\ ft^3$

Then the empty weight:

$W_e = 0.5 \times 658722.2$

$W_e = 329361.1 lb$ ...[3.15]

3.3 Estimation of the critical performance parameters

3.3.1 Maximum lift coefficient

An initial choice of the airfoil section for our airplane design has been made as follows : at tip section an MH-62 , at root an MH-61 and at the body an MH-93, from historical designs of models (tailless aircrafts) .

For wing:

Average (c_l) max $= \frac{1.1880+1.3343}{2} = 1.26$

For body:

$((c_l)$max)root =1.1880 (($c_l)$max)body =1.3783

Average (c_l) max $= \frac{1.1880+1.3783}{2} = 1.28$

Average (c_l)max with 45° flap deflection =1.26+0.9=2.16

Total average (c_l) max$=\frac{2.16+1.28}{2} = 1.72$

Finally to account for three-dimensional effect of finite aspect ratio, raymer suggests that, for finite wings and body with aspect ratio greater than 5:

$(C_L)max = 0.9\ (c_l)max = 0.9 \times 1.72$

$(C_L)max = 1.548$ ...[3.16]

This $(C_L)max$ for complete airplane

3.3.2 Wing loading W/S

in most airplane designs, wing loading determined by consideration of V_{stall} and landing distance.

By stall velocity:

$V_{stall} = \sqrt{\frac{2}{\rho} \times \frac{W}{S} \times \frac{1}{c_l)max}}$ ...[3.17]

$\frac{W}{S} = \frac{1}{2}\rho {V_{stall}}^2 (C_L)max$

We make the calculation at sea level, where $\rho_\infty = 0.002377\ slug/ft^2$

$\frac{W}{S} = \frac{1}{2} \times 0.002377 \times 250^2 \times 1.548$

$\frac{W}{S} = 115\ lb/ft^2$ ...[3.18]

By landing distance:

Average velocity during flare

V_f=1.23V_{stall} ...[3.19]

$= 1.23 \times 250$

$V_f = 307.5\ ft/s$ ...[3.20]

The flight path radius during flare

$R = \frac{{V_f}^2}{2g}$...[3.21]

$= \frac{307.5^2}{0.2 \times 32.2}$

$R = 14682.6\ ft$ ...[3.22]

The approach angle $\theta_a \leq 3$ for transport aircraft that is θ_a=3

The flare height

$h_f = R(1 - \cos\theta_a)$...[3.23]

$= 14682.6(1 - \cos 3)$

$h_f = 20.12\ ft$ ...[3.24]

The approach distance required to clear a 50 ft obstacle

$S_a = \frac{50 - h_f}{\tan\theta_a}$...[3.25]

$$= \frac{50 - 20.12}{\tan 3}$$

$S_a = 570.14\ ft$...[3.26]

The flare distance

$S_f = R\sin\theta_a$ ...[3.27]

$= 14682.6 \sin 3$

$S_f = 768.42\ ft$ ...[3.28]

Ground roll

$S_g = jN\sqrt{\frac{2}{\rho_\infty} \times \frac{W}{S} \times \frac{I}{(C_L)max}} + \frac{j^2 \times \frac{W}{S}}{g\rho(C_L)max\mu_r}$ ...[3.29]

J=1.15 for commercial airplanes

$\mu_r = 0.4$

N= 3s (depend on pilot technique 1_3 s)

$$S_g = 1.15 \times 3\sqrt{\frac{2}{0.002377} \times \frac{W}{S} \times \frac{1}{1.548}} + \frac{1.15^2 \times \frac{W}{S}}{32.2 \times 0.002377 \times 1.548 \times 0.4}$$

$$S_g = 80.43\sqrt{\frac{W}{S}} + 27.9\frac{W}{S}$$

Since the allowable landing distance is specified in requirements as 6561 ft, and we have previously estimated that $S_a = 570.14\, ft\ and\ S_f = 768.42\, ft$

The allowable value for S_g Is:

$$S_g = 6561 - 570.14 - 768.42$$

$$S_g = 5222.44\, ft \quad ...[3.30]$$

$$5222.44 = 80.43\sqrt{\frac{W}{S}} + 27.9\frac{W}{S}$$

Then:

$$\frac{W}{S} = \ lb/ft^2 \quad ...[3.31]$$

We choose

$$\frac{W}{S} = 115\, lb/ft^2 \quad ...[3.18]$$

The wing area:

$$S = \frac{W_0}{\frac{W}{S}} = \frac{658722.2}{115}$$

$$S = 5728\, ft^2 \quad ...[3.32]$$

3.3.3 Thrust to weight ratio

The value of T/W is determined by takeoff distance, rate of climb, and maximum velocity.

By takeoff distance:

$$S_g = \frac{1.21 \times \frac{W}{S}}{g \times \rho \times (C_L)max \times \frac{T}{W}} \quad ...[3.33]$$

Where the $C_{l_{max}}$ is value with the flaps only partially extended, we assume a flap deflection of 20° for takeoff . the $\Delta C_{l_{max}}$ for a 45° flap deflection is 0.9. assuming a linear variation of $\Delta C_{l_{max}}$ with flap deflection angle, we have for takeoff:

$$\Delta C_{l_{max}} = 0.9 \times \frac{25}{45} = 0.5$$

The average$(c_l)max$ With 20 degrees flap deflection:

$$(c_l)max = 1.26 + 0.5 = 1.76$$

Total average (c_l) max=$\frac{1.76+1.28}{2} = 1.52$

$$(C_L)max = 0.9 \times (c_l)max = 0.9 \times 1.52 = 1.368$$

And we have:

$$S_g = \frac{1.21 \times 115}{32.2 \times 0.002377 \times 1.368 \times \frac{T}{W}}$$

$$V_{stall} = \sqrt{\frac{2}{\rho} \times \frac{W}{S} \times \frac{1}{c_l)max}} \qquad ...[3.17]$$

$$= \sqrt{\frac{2 \times 115}{0.002377 \times 1.368}}$$

$$V_{stall} = 266 \frac{ft}{s} \qquad ...[3.34]$$

Flight bath radius

$$R = \frac{6.96 \times {V_{stall}}^2}{g}$$

$$= \frac{6.96 \times 266^2}{32.2}$$

$$R = 15293.8\, ft \qquad ...[3.35]$$

Flight bath angle

$$\theta_{OB} = \cos^{-1}\left(1 - \frac{h_{OB}}{R}\right)$$

$$= \cos^{-1}\left(1 - \frac{50}{15293.8}\right)$$

$$\theta_{OB} = 4.63\, degree \qquad ...[3.36]$$

Airborne distance

$$S_a = R \sin \theta_{OB}$$

$$= 15293.8 \times \sin 4.63$$

$$S_a = 1234.5\ ft \qquad \text{...[3.37]}$$

Combining the two equation:

$$S_g + S_a = 8202 = \frac{1329}{\frac{T}{W}} + 1234.5$$

$$\frac{T}{W} = \frac{1329}{8202 - 1234.5}$$

$$\frac{T}{W} = 0.19 \qquad \text{...[3.38]}$$

This is the value of required T/W at velocity:

$$V_\infty = 0.7V_{lo} = 0.7(1.1V_{stall}) \qquad \text{...[3.39]}$$

$$= 0.7 \times 1.1 \times 266$$

$$V_\infty = 204.82\ ft/s \qquad \text{...[3.40]}$$

Thrust required:

$$T_R = \frac{T}{W} \times W = 0.1907 \times 658722.2$$

$$T_R = 125618.3\ lb \qquad \text{...[3.41]}$$

by maximum speed at mid cruise:

from statistics:

-$C_{fe} = 0.003$ $\qquad \frac{S_{wet}}{S} = 2.4$

Then:

$$C_{D,0} = C_{fe} \times \frac{S_{wet}}{S} = 0.003 \times 2.4$$

$$C_{D,0} = 0.0072 \qquad \text{...[3.42]}$$

E=0.85

$$\frac{L}{D})_{max} = \sqrt{\frac{1}{4KC_{D,0}}} = \sqrt{\frac{1}{4 \times K \times 0.0072}} = 21$$

K=0.0787…[3.43]

AR=$\frac{1}{\pi \times e \times k} = \frac{1}{\pi \times 0.85 \times 0.0787}$

$\mathrm{AR} = 4.76$ …[3.44]

In level flight, T=D

we assume that the specified V_{max} associated with level flight at 35000 ft

T=D=$\frac{1}{2}\rho V^2 S C_{D,0} + \frac{2KS}{\rho V^2}(\frac{W}{S})^2$ …[3.45]

$$\frac{T}{W} = \frac{1}{2}\rho V^2 \frac{C_{D,0}}{W/_S} + \frac{2K}{\rho V^2}\frac{W}{S}$$

$$\frac{W_2}{W_0} = \frac{W_1}{W_0}\frac{W_2}{W_1} = 0.97 \times 0.985$$

$\frac{W_2}{W_0} = 0.955$…[3.46]

$$W_2 = 0.955 \times W_0 = 0.955 \times 567000$$

$W_2 = 541485\ lb$…[3.47]

Mid cruise weight W_{MC}

$\frac{W_{MC}}{W_2} = \frac{1}{2}\left(1 + \frac{W_3}{W_2}\right)$ …[3.48]

$$= \frac{1}{2}(1 + 0.753)$$

$\frac{W_{MC}}{W_2} = 0.8765$ …[3.49]

$$W_{MC} = 0.8765 \times 541485$$

$W_{MC} = 474611.6\ lb$ …[3.50]

$$\frac{W_{MC}}{S} = \frac{474611.6}{2465.2}$$

$\frac{W_{MC}}{S} = 192.52\ lb/ft$ …[3.51]

$$\frac{T}{W_{MC}} = \frac{1}{2}\rho V^2 \frac{C_{D,0}}{W_{MC}/_S} + \frac{2K}{\rho V^2}\frac{W_{MC}}{S} \qquad ...[3.52]$$

$$\frac{T}{W_{MC}} = \frac{1}{2} \times 0.0007382 \times 872.24^2 \times \frac{0.0072}{192.52} + \frac{2 \times 0.0787}{0.0007382 \times 872.24^2} \times 192.52$$

$$= 0.0645$$

$$T = 0.0645 \times 474611.6$$

$$T = 30612.35\ lb \qquad ...[3.53]$$

by rate of climb:

$$(^R/_C)_{max} = \left(\frac{\frac{W}{S}Z}{3\rho_\infty C_{D,0}}\right)^{\frac{1}{2}} \left(\frac{T}{W}\right)^{\frac{3}{2}} \left(1 - \frac{Z}{6} - \frac{3}{2(\frac{T}{W})^2(\frac{L}{D})_{max}^{\ \ 2}Z}\right) \qquad ...[3.54]$$

$$Z = 1 + \sqrt{1 + \frac{3}{\left(\frac{T}{W}\right)^2 \left(\frac{L}{D}\right)_{max}^{\ \ 2}}} \qquad ...[3.55]$$

$$Z = 1 + \sqrt{1 + \frac{1}{147(\frac{T}{W})^2}}$$

$$58.33 = (\times Z)^{\frac{1}{2}} \left(\frac{T}{W}\right)^{\frac{3}{2}} \left(1 - \frac{Z}{6} - \frac{1}{Z}\right)$$

3.4 Configuration layout

3.4.1 Wing configuration

We have two considerations, the geometric shape of the wing and it is location relative to the fuselage. First we consider the shape.

Wing geometry is describe by (a) aspect ratio, (b) wing sweep, (c) taper ratio, (d) variation of airfoil shape and thickness along the span, and (e) geometric twist.

1- **aspect ratio**

$$AR = \frac{b^2}{s} \qquad ...[3.56]$$

AR= 4.76 S=2465.2

$$b = \sqrt{AR \times S} \qquad ...[3.57]$$

$$= \sqrt{4.76 \times 5728}$$

$$b = 165.12\ ft \quad \text{...[3.58]}$$

2- wing sweep

we assume that 50%chord sweep=32°

3- taper ratio

From fig 2.39 , see that for our aspect ratio of 4.76 , the minimum value of δ= 0.002 occurs at taper ratio 0.3

$$\acute{C} = \frac{b}{AR} \quad \text{...[3.59]}$$

$$= \frac{165.12}{4.76}$$

$$\acute{C} = 34.7ft \quad \text{...[3.60]}$$

$$\acute{C} = \frac{2}{3} C_r \left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right) \quad \text{...[3.61]}$$

$$C_r = 48.68\ ft \quad \text{...[3.62]}$$

$$\lambda = \frac{C_t}{C_r} \quad \text{...[3.63]}$$

$$C_t = 14.6\ ft \quad \text{...[3.64]}$$

3.4.2 Fuselage configuration

Nose and Cockpit - Front Fuselage:

for nose:

$$l_{nose} = 1m = 3.28ft \quad \text{...[3.65]}$$

For the cockpit length (lcockpit), standards have been prescribed by Raymer (Reference 1.11,chapter 9). lcockpit for the two member crew is chosen as 100 inches (2.5 m)

$$l_{cockpit} = 2.5m = 8.2\ ft \quad \text{..[3.66]}$$

Passenger Cabin Layout

Seating arrangement

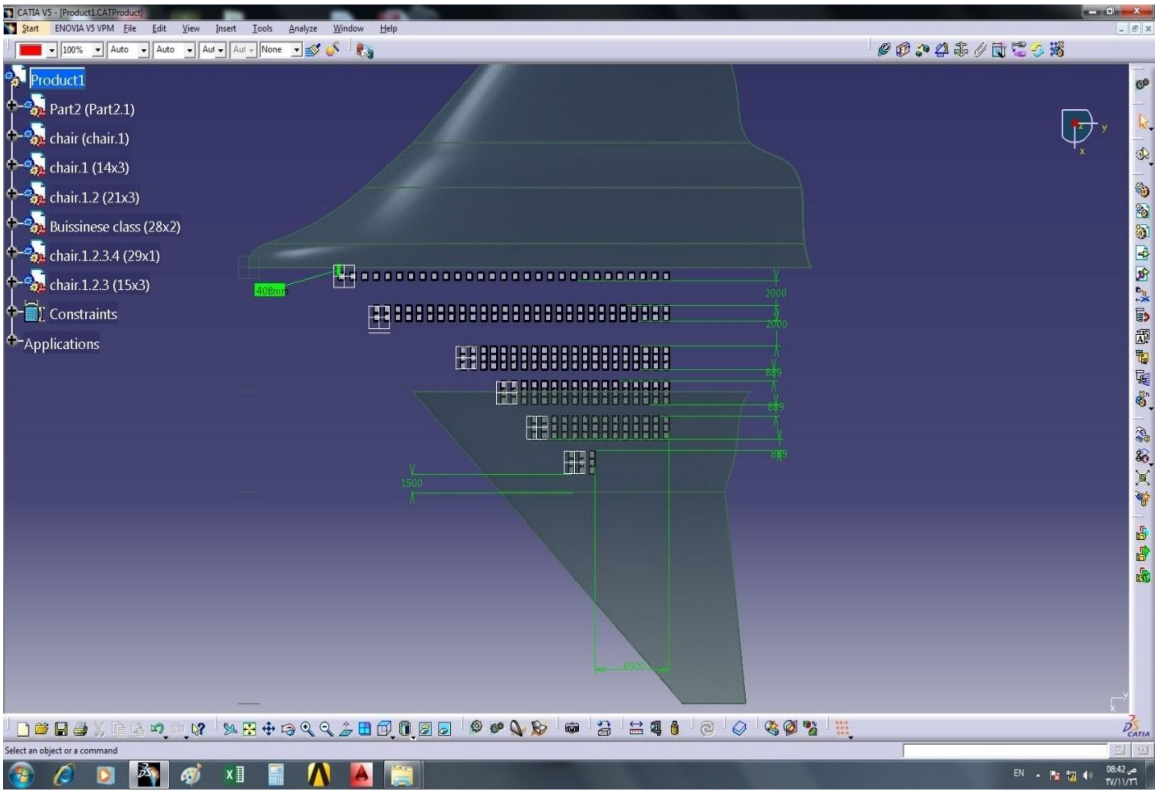

Figure 3.23 seating arrangement

Cabin length

The design contains three compartments (first class , business and economy). The total number of seats is (500) . its distributed as 58 seats in the first class , 112 seats in the business class and 330 seats in the economy class (228 seats in the body and 102 seats in the wing-body section) .

Cabin parameters are chosen based on standards for similar airplanes. The various parameters are as follows:

Table 3.2 fuselage parameters

Parameter	First class	Business class	Economy class
Seat pitch (in inches)	40	37	35
Seat width (in inches)	35	25	25
Aisle width (in inches)	79 + 6 from A\C center line	79	35 + 59 from the root
Seats abreast	2	4	12 in body+12 in wing

Table 3.3 cabin length estimation

Class	No.of seats	No.of rows	Seat pitch (in)	Cabin length (ft)
Frist class	58	29	40	96.67

Fuselage total length:

The provision of service modules and the 'wasted' space adjacent to the doors will add about 4 m (13.12 ft) to the cabin length

the total fuselage length = 3.28+8.2+96.67+13.12=121.27 ft ...[3.67]

Cabin width

From the seating arrangement done by CATIA

Total cabin width=118.11 ...[3.68]

3.4.3Resulting layout

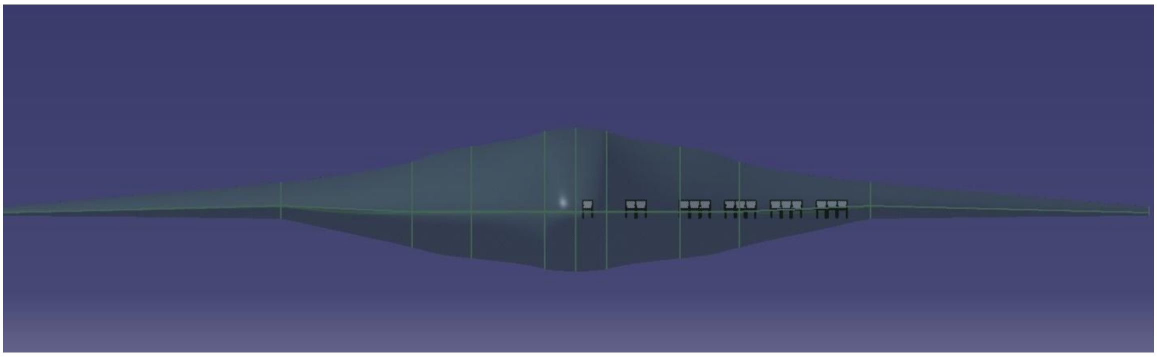

Figure 3.24 front view

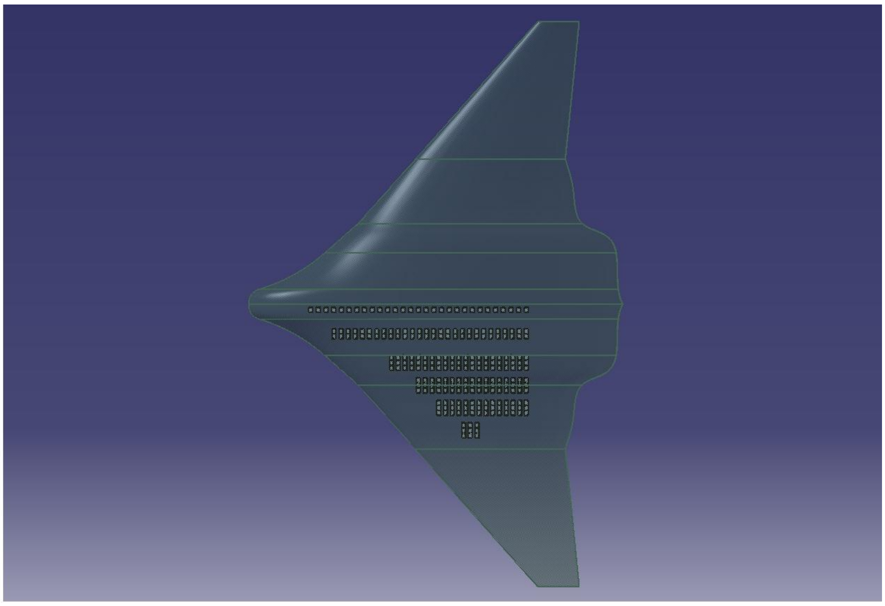

Figure 3.25 top view

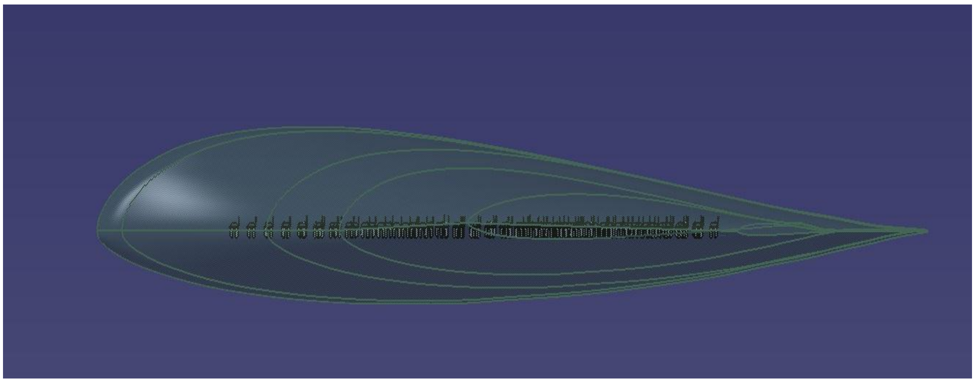

Figure 3.26 side view

configuration layout

3.4.4 Results From the drawing:

Table 4 Results from the drawing

Root chord of fuselage	161.6 ft
Outer fuselage chord	97 ft
Wing root chord	64.6 ft
Wing tip chord	20.3 ft
Length from fuselage center to outer	32.8 ft
Length from fuselage outer to wing root chord	26.2 ft
Length from wing root chord to wing tip chord	55.8 ft
Total aircraft span	229.66 ft
Total aircraft length	164 ft

in order for the aircraft to be able to carry 500 passengers the aircraft span has to be

b= 229.66 ft

and from that :

$$S = \frac{b^2}{AR} \qquad \text{...[3.56]}$$

$$= \frac{229.66^2}{4.76}$$

$$S = 11080.6\ ft^2 \qquad \text{...[3.69]}$$

$$\frac{W}{S} = 115 = \frac{W}{11080.6}$$

$$W = 1274269\ lb \qquad \text{...[3.70]}$$

$$\frac{T}{W} = 0.19 = \frac{T}{1274269}$$

$$T = 242111.11\ lb \qquad \text{...[3.71]}$$

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - 0.32 - 0.5} = 1274269\ lb$$

$$W_{crew} + W_{payload} = 229368.42\ lb \qquad \text{...[3.72]}$$

$$\frac{W_f}{1274269} = 0.32$$

$$W_f = 407766.08\ lb \qquad \text{...[3.73]}$$

$$W_e = 0.5 \times 1274269$$

$$W_e = 637134.5\ lb \qquad ...[3.74]$$

$$Ć = \frac{b}{AR} \qquad ...[3.59]$$

$$= \frac{229.66}{4.76}$$

$$Ć = 48.25ft \qquad ...[3.75]$$

$$Ć = \frac{2}{3} C_r \left(\frac{1+\lambda+\lambda^2}{1+\lambda}\right) \qquad ...[3.61]$$

$$C_r = 67.686\ ft \qquad ...[3.75]$$

$$\lambda = \frac{C_t}{C_r}$$

$$C_t = 20.3\ ft \qquad ...[3.76]$$

3.4.5Engine Selection

CF6-80C2

Currently certified on 12 wide body aircraft models and with 16 ratings, the CF6-80C2 has accumulated over 200 million flight hours in service.

Figure 3.27 CF6-80C2

Specification

Type	High bypass turbofan engine
Max.thrust	63,500lb
Length	168 in
Airflow	1,790 lb/sec
Bypass ratio	5.3
Pressure ratio	32.6
Weight	9859 lb

3.4.6Center of gravity location: First estimate

Is calculated by summing moments about the nose and dividing by the sum of the weights the result is

$$\bar{x} = \frac{(W_{crew} + W_{payload}) \times L_1 + W_{engine} \times L_2}{W_{crew} + W_{paylod} + W_{engine}}$$

$$= \frac{229368.42 \times 82 + 4 \times 13802.6 \times 171}{229368.42 + 4 \times 13802.6}$$

$$= \frac{28249188.84}{284578.82}$$

$$\bar{x} = 99.3 ft \qquad ...[3.77]$$

In the above calculation the weight of the installed engine is taken as 1.4 times the given dry weight of 9859 lb as suggested by Raymer

Usual design procedure calls for locating the wing relative to the fuselage such that the mean aerodynamic center of the wing is close to the c.g of the airplane

Raymer suggests that we estimate the weight of wing by multiplying the plan form area by 2.5

$$W_{wing} = 2.5 \times 11080.5$$

$$W_{wing} = 27701.5\ lb \qquad ...[3.78]$$

We also assume that the wing aerodynamic center is 25% of the mean aerodynamic chord from the leading edge, and that the wing center of gravity is at 40% of the mean aerodynamic chord

$$m.a.c = 0.25 \times 48.25 = 12.063\ ft \qquad ...[3.79]$$

$$c.g_w = 0.4 \times 48.25 = 19.3 \qquad ...[3.80]$$

$$\bar{x} = \frac{28249188.84 + 27701.5 \times (99.3 + 7.237)}{284578.82 + 27701.5}$$

$$\bar{x} = 99.9\ ft \qquad ...[3.81]$$

3.5 A better weight estimate

Raymer gives an approximate weight buildup for a general aviation airplane as follows

$$wing\ weight = 2.5 S_{exposed\ wing\ planform} \qquad ...[3.82]$$

$$fuselage\ weight = 1.4 S_{watted\ area} \qquad ...[3.83]$$

$$installed\ engine\ weight = 1.4(engine\ weight) \qquad ...[3.84]$$

$$landing\ gear\ weight = 0.057 W_0 \qquad ...[3.85]$$

$$all\ else\ empty = 0.1 W_0 \qquad ...[3.86]$$

From CATIA we get that

$$S_{exposed\ wing\ planform} = 9452.6\ ft^2 \qquad ...[3.87]$$

$$S_{watted\ area} = 27810.1\ ft^2 \qquad ...[3.88]$$

Then we have

$$wing\ weight = 2.5 \times 9452.6 = 23631.5\ lb ...[3.89]$$

$$fuselage\ weight = 1.4 \times 27810.1 = 38934.14\ lb ...[3.90]$$

$$installed\ engine\ weight = 1.4 \times 4 \times 9859 = 55210.4\ lb \qquad ...[3.91]$$

$$landing\ gear\ weight = 0.057 \times 1274269 = 72633.33\ lb \qquad ...[3.92]$$

$$all\ else\ empty = 0.1 \times 1274269 = 127426.9\ lb \qquad ...[3.93]$$

$$total\ empty\ weight$$

$W_e = 317836.27\ lb$ ...[3.94]

$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty}$
$= 229368.42 + 407766.08 + 317836.27$

$W_0 = 954971\ lb$ ...[3.95]

And from that:

$W_{fuel} = 0.32 \times 954971$

$W_{fuel} = 305590.72\ lb$ ...[3.96]

3.6 Performance analysis

$\frac{W}{S} = 86.2\ lb/ft^2$ ...[3.97]

$\frac{T}{W} = 0.25$ ...[3.98]

$C_{D,0} = 0.0072$ ...[3.42]

$K = 0.0787$ ...[3.43]

$(C_L)max = 1.548$ ...[3.16]

$(\frac{L}{D})_{max} = 21$ ...[3.4]

3.6.1 Thrust required and thrust available

Thrust required and thrust available calculated at altitude 43000 ft

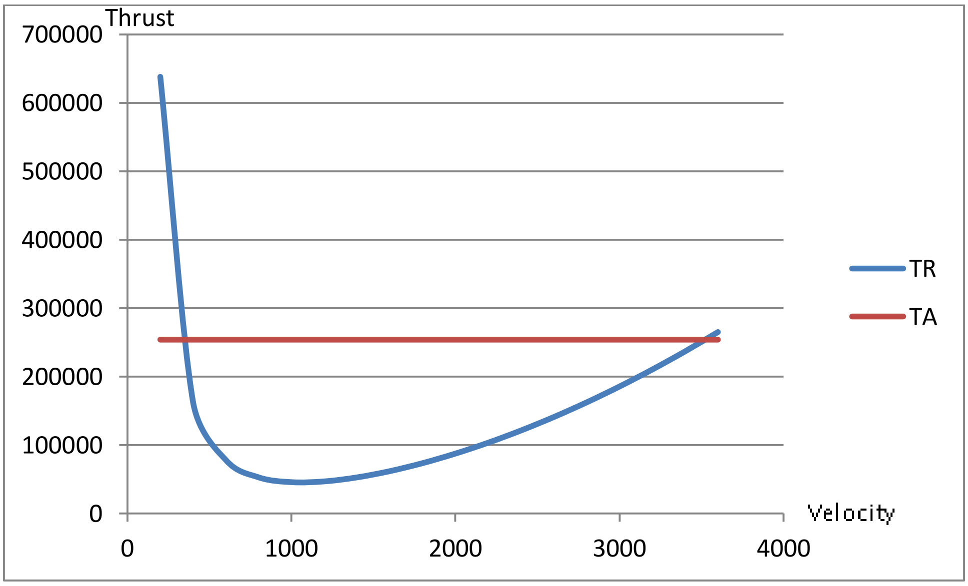

Figure 3.28 thrust required and thrust available at 43000 ft

3.6.2 Power required and power available

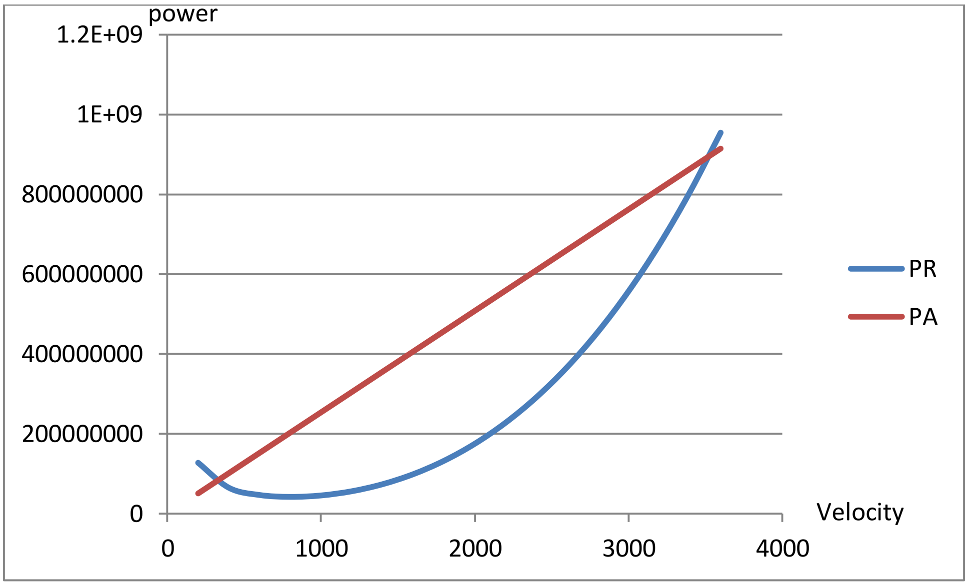

Figure 3.29 Power required and power available at 43000 ft

3.6.3 Stalling speed

$$V_{stall} = \sqrt{\frac{2}{\rho} \times \frac{W}{S} \times \frac{1}{c_l)max}} \qquad \text{...[3.17]}$$

$$V_{stall} = \sqrt{\frac{2 \times 86.2}{0.002377 \times 1.548}}$$

$$V_{stall} = 216.5\frac{ft}{s} \qquad \text{...[3.99]}$$

3.6.4 Rate of climb

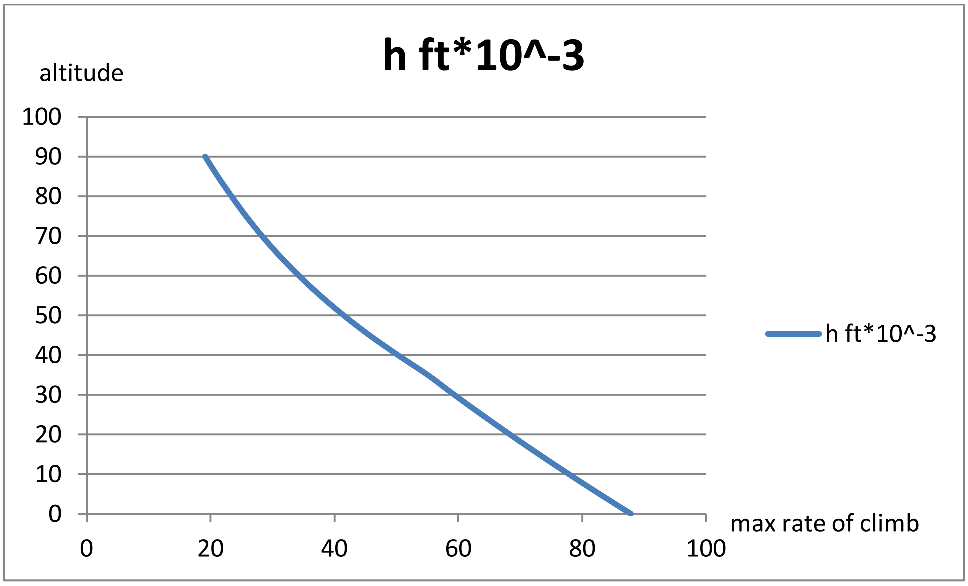

Figure 3.30 maximum rate of climb as a function of altitude

the variation of maximum rate of climb with altitude is shown in figure (3.28) where the weight at each altitude assumed to be $W_0 = 954971\ lb$,at sea level (R/C)max =8786 ft/min. this far exceeds the required (R/C)max =3500 ft/min ,hence our plane design exceeds the specification.

3.6.5 Range

Since we are assuming the same aerodynamic characteristics for the airplane figure (3.28)

As we have used during the earlier part of this chapter the range also stays the same. For a range of 5500 nmi $\frac{W_f}{W_0} = 0.32$ as calculated however because of the lighter gross weightW_f is smaller we have already calculated the new fuel weight to be $305590.72\ lb$ down from our first estimate of $407766.08\ lb$, hence our aircraft design meet the specification for range of 5500nmi and it does this with smaller fuel load than had previously been calculated.

3.6.6 Landing distance

Average velocity during flare

V_f=1.23V_{stall} ...[3.19]

$$= 1.23 \times 216.5$$

$V_f = 266.3\ ft/s$ ...[3.100]

The flight path radius during flare

$R = \frac{{V_f}^2}{2g}$ ...[3.21]

$$= \frac{266.3^2}{0.2 \times 32.2}$$

$R = 11011.8 ft$ ...[3.101]

The approach angle $\theta_a \leq 3$ for transport aircraft that is θ_a=3

The flare height

$h_f = R(1 - \cos\theta_a)$ ...[3.23]

$$= 11011.8(1 - \cos 3)$$

$h_f = 15.1 ft$ ...[3.102]

The approach distance required to clear a 50 ft obstacle

$S_a = \frac{50 - h_f}{\tan\theta_a}$...[3.25]

$$= \frac{50 - 15.1}{\tan 3}$$

$S_a = 666\ ft$ ...[3.103]

The flare distance

$S_f = R \sin\theta_a$ ...[3.27]

$$= 11011.8 \sin 3$$

$S_f = 576.3\ ft$ ...[3.104]

Ground roll

$$S_g = jN\sqrt{\frac{2}{\rho_\infty} \times \frac{W}{S} \times \frac{I}{(C_L)max}} + \frac{j^2 \times \frac{W}{S}}{g\rho(C_L)max\mu_r}$$ ...[3.29]

J=1.15 for commercial airplanes

$\mu_r = 0.4$

N= 3s (depend on pilot technique 1_3 s)

$$S_g = 1.15 \times 3\sqrt{\frac{2}{0.002377} \times 86.2 \times \frac{1}{1.548}} + \frac{1.15^2 \times 86.2}{32.2 \times 0.002377 \times 1.548 \times 0.4}$$

$S_g = 3152\ ft$ ...[3.105]

$\text{total landing distance} = S_g + S_f + S_a = 3152 + 576.3 + 666$

$\text{total landing distance} = 4394.3\ ft$ ...[3.106]

3.6.7 Takeoff distance

$$S_g = \frac{1.21 \times \frac{W}{S}}{g \times \rho \times (C_L)max \times \frac{T}{W}}$$ ...[3.33]

Where the $C_{l_{max}}$ is value with the flaps only partially extended, we assume a flap deflection of 20°for takeoff . the $\Delta C_{l_{max}}$ for a 45° flap deflection is 0.9. assuming a linear variation of $\Delta C_{l_{max}}$ with flap deflection angle, we have for takeoff:

$$\Delta C_{l_{max}} = 0.9 \times \frac{25}{45} = 0.5$$

The average$(c_l)max$ With 20 degrees flap deflection:

$(c_l)max = 1.26 + 0.5 = 1.76$

Total average (c_l) max=$\frac{1.76+1.28}{2} = 1.52$

$(C_L)max = 0.9 \times (c_l)max = 0.9 \times 1.52 = 1.368$

And we have:

$$S_g = \frac{1.21 \times 86.2}{32.2 \times 0.002377 \times 1.368 \times 0.25}$$

$S_g = 3985\ ft$...[3.107]

$$V_{stall} = \sqrt{\frac{2}{\rho} \times \frac{W}{S} \times \frac{1}{c_l)max}} \qquad \text{...[3.17]}$$

$$= \sqrt{\frac{2 \times 86.2}{0.002377 \times 1.368}}$$

$V_{stall} = 230.3\ ft/s$
...[3.108]

Flight bath radius

$$R = \frac{6.96 \times {V_{stall}}^2}{g}$$

$$= \frac{6.96 \times 230.3^2}{32.2}$$

$R = 11464.1\ ft.$
...[3.109]

Flight bath angle

$$\theta_{OB} = \cos^{-1}\left(1 - \frac{h_{OB}}{R}\right) = \cos^{-1}\left(1 - \frac{50}{11464.1}\right)$$

$\theta_{OB} = 5.35\ degree$
...[3.110]

Airborne distance

$$S_a = R \sin \theta_{OB} = 11464.1 \times \sin(5.35)$$

$S_a = 1069 ft$
...[3.111]

Combining the two equation:

$$total\ takeoff\ distance = S_g + S_a = 3985 + 1069$$

$total\ takeoff\ distance = 5054\ ft$
...[3.112]

Chapter 4 : RESULTS &DISCUSION

4.1 Results

4.1.1 Results from calculation

Table 4.5 results from calculations

parameter	value
W_0	$658722.2\ lb$
W_f	$210791.1\ lb$
Tank capacity	$37374.3 gal$
Tank volume	$5008.16\ ft^3$
W_e	$329361.1 lb$
$(C_L)max$	1.548
$\frac{W}{S}$	$115\ lb/ft^2$
S	$5728\ ft^2$
$\frac{T}{W}$	0.19
T_R	$125618.3\ lb$
$C_{D,0}$	0.0072
$(\frac{L}{D})_{max}$	21
k	0.0787
AR	4.76
b	$165.12\ ft$
λ	0.3
Ć	$34.7 ft$
C_r	$48.68\ ft$
C_t	$14.6\ ft$

4.1.2 Results from CATIA

Table 4.6 results from CATIA

C_t	20.3 ft
C_r	64.6 ft
Ć	$48.25 ft$
b	229.66 ft
T_R	$242111.11\ lb$
W_e	$637134.5\ lb$
W_0	$1274269\ lb$
W_f	$407766.08\ lb$
L	164 ft
$\bar{x}$	$99.9\ ft$

4.1.3 Results after better weight estimation has been achieved

Table 4.7 after better weight estimation results

W_e	$317836.27\ lb$
W_0	$954971\ lb$
W_f	$305590.72\ lb$
$\frac{W}{S}$	$86.2\ lb/ft^2$
$\frac{T}{W}$	0.25
V_{stall}	$216.5 \frac{ft}{s}$
(R/C)max	8786 ft/min
Landing distance	$4394.3\ ft$
Takeoff distance	$5054\ ft$

4.1.4 Seating arrangement

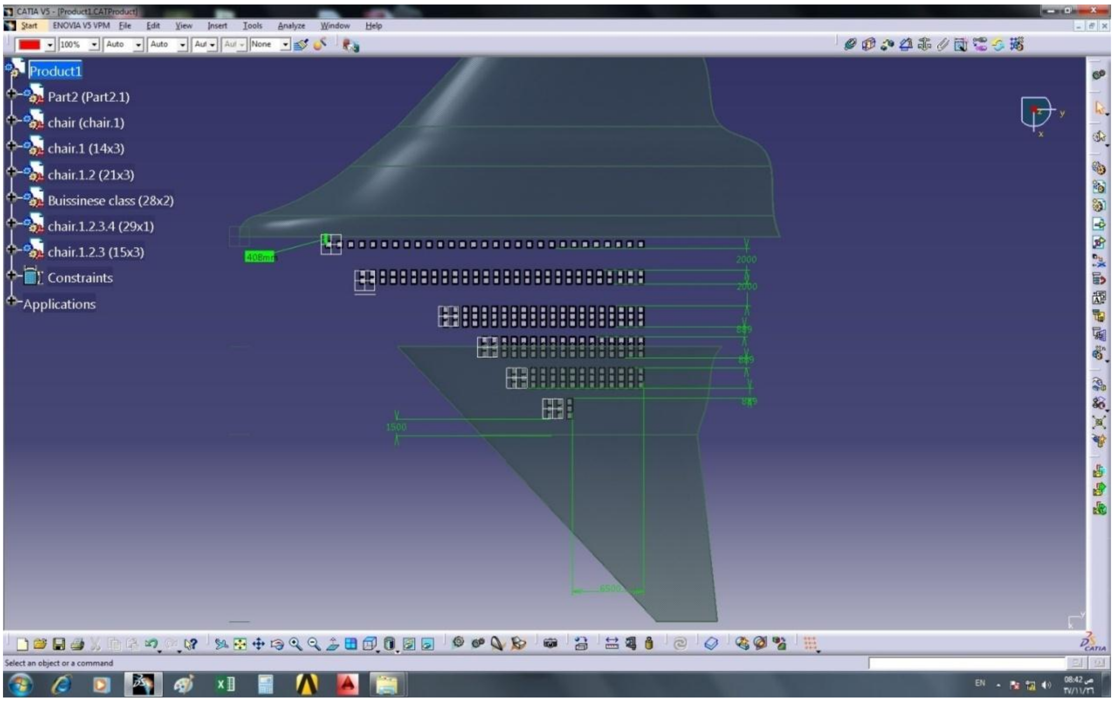

Figure 4.31 seating arrangement

4.1.5 Resulting layout

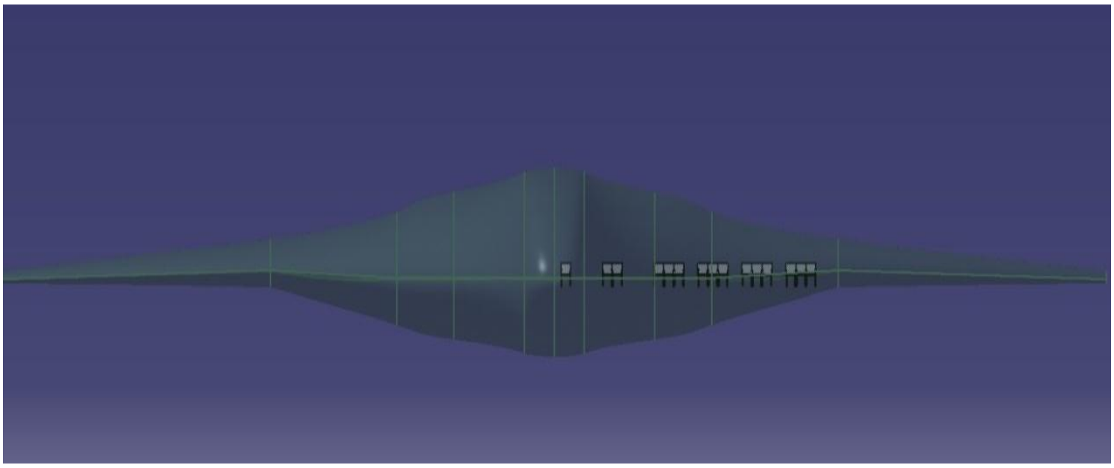

Figure 4.32 front view

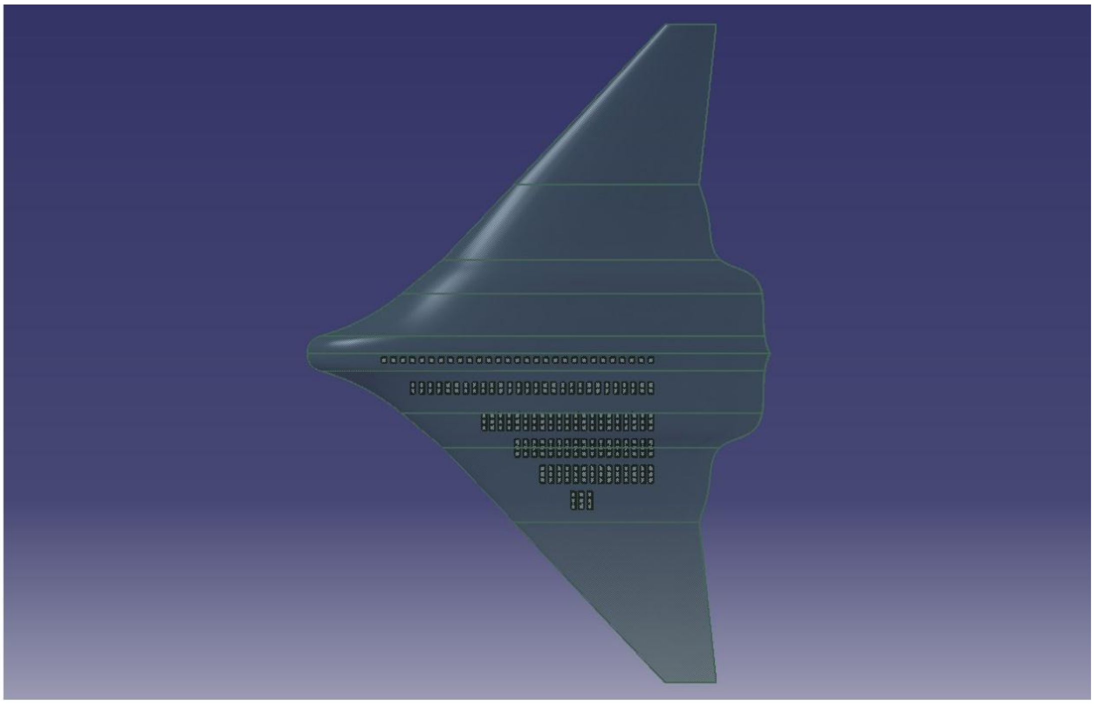

Figure 4.33 top view

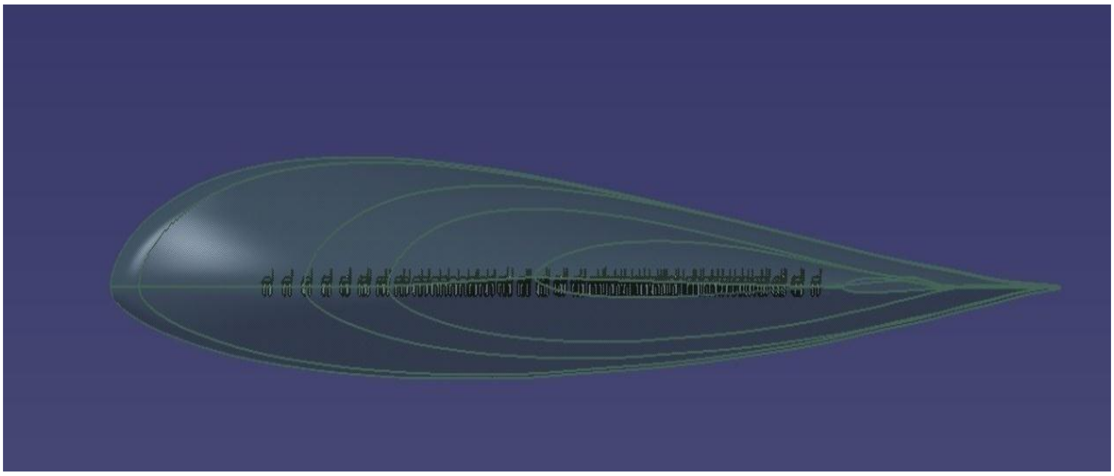

Figure 4.34 side view

configuration layout

4.2 Discussion

- One of the main design goals of the BWB is to reduce the total takeoff weight , and as shown in the result the weight of 500 blended wing body is less than conventional aircrafts of the same capacity.
- Calculations of maximum lift coefficient (CLmax) for a blended wing body is very complicated and can not be obtained by applying conventional aircrafts formula , the method we used here is approximate.
- W/S is similar to the value of aircrafts of the same weight
- T/W value is acceptable to the total weight of aircraft
- The aircraft aspect ratio was somehow low (AR) because of the small wing span while the wing area is quite large.
- The chosen Airfoils thickness was small we had to scale it up for the aircraft in order to be able to carry the number of passengers set in the requirement
- The value of wing span (b) estimated in the initial calculation was too small , it has been replaced with the value obtained from the drawing.
- The change in wing span value has led to other changes in wing area (S) , weight (W) , and thrust (T) ,as all of their values has increased . as well as Cmean , Cr and Ct
- To obtain the thrust required we chose four high bypass turbofan engines (CF6-80C2) same as the one used in Boeing 787, each engine is of 63,500lb maximum thrust.

Centre of gravity first estimation

$$\bar{x} = \frac{\left(W_{crew} + W_{payload}\right) \times L_1 + W_{engine} \times L_2}{W_{crew} + W_{paylod} + W_{engine}} = 99.3ft$$

The location of CG has been estimated regardless the wing existence .

We assumed according to Raymer that:

-the wing aerodynamic centre is 25% often mean aerodynamic chord.

-wing center of gravity is at 40% of the mean aerodynamic chord .

$$m.a.c = 0.25 \times 48.25 = 12.063\ ft$$

$$\bar{x} = 99.9\ ft$$

The value of CG is somehow large , and that's because the landing gear location and weight isn't considered in the calculation.

The wing exposed planform area & fuselage wetted area values were obtained for the drawing , to calculate the better weight estimation.

The value of the aircraft empty weight has dropped from ($637134.5\ lb$) to ($317836.27\ lb$) and hence the maximum take-off weight dropped to ($954971\ lb$).

Performance analysis:

- From the power required & power available curve we notice , the maximum and minimum speeds are (3500 ft/sec , 360 ft/sec) respectively.

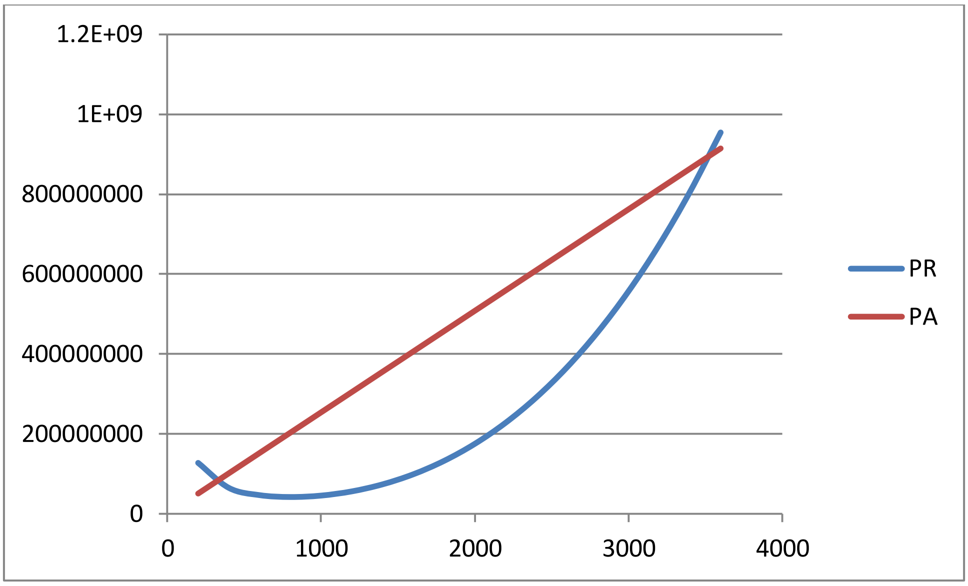

Figure 4.35 Power required and power available at 43000 ft

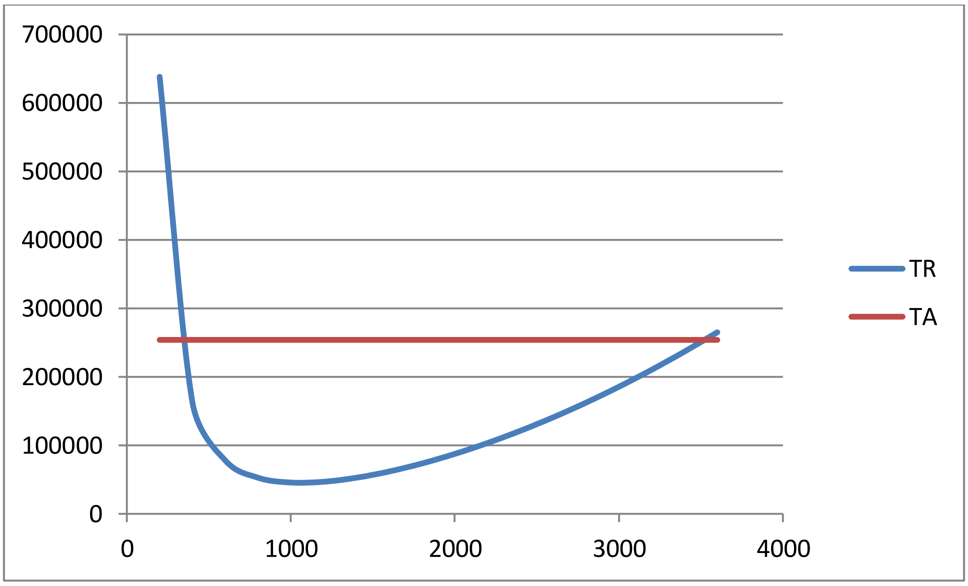

Figure 4.36 thrust required and thrust available at 43000 ft

- The stall speed is found to be 216.5 m/s after the better weight estimation , which has lessened from 250 m/s estimated in the initial calculation.

- the variation of maximum rate of climb with altitude is shown in figure (4.37) where the weight at each altitude assumed to be $W_0 = 954971\ lb$,at sea level (R/C)max =8786 ft/min. this far exceeds the required (R/C)max =3500 ft/min ,hence our plane design exceeds the specification.

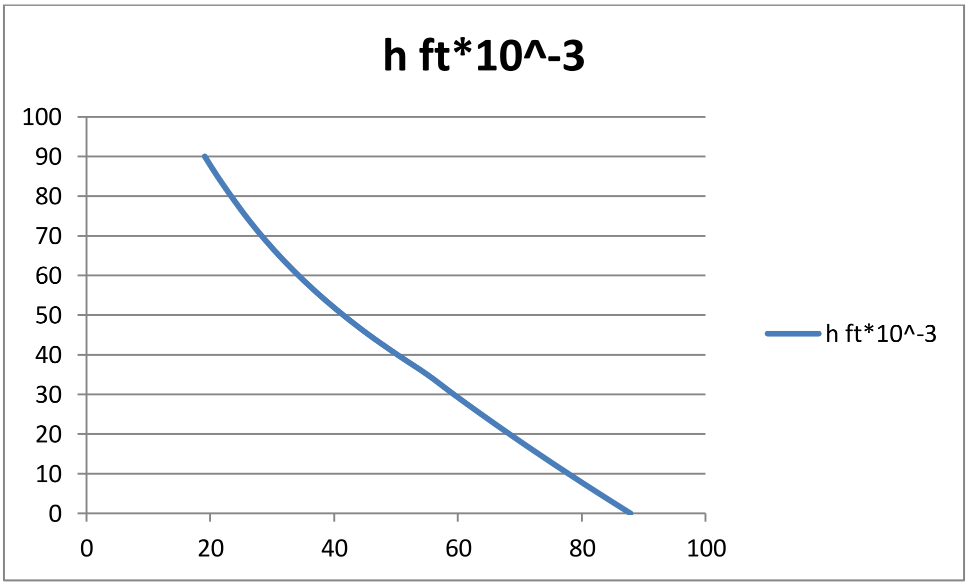

Figure 4.37 maximum rate of climb as a function of altitude

- Since we are assuming the same aerodynamic characteristics for the airplane figure (4.37)

As we have used during the earlier part of this chapter the range also stays the same. For a range of 5500 nmi $\frac{W_f}{W_0} = 0.32$ as calculated however because of the lighter gross weightW_f is smaller we have already calculated the new fuel weight to be $305590.72\ lb$ down from our first estimate of $407766.08\ lb$, hence our aircraft design meet the specification for range of 5500nmi and it does this with smaller fuel load than had previously been calculated.

- The take-off & landing distances have shown a major reduction in their values from the ones set in the requirements to ($5054\ ft$, $4394.3\ ft$).
- The value of the parameters obtained after the better weight estimated serve to optimize the aircraft performance.

Chapter 5 : CONCLUSIONS & RECOMMENDATIONS

5.1 conclusion

The aim of this thesis was to study the design concept of a blended wing body aircraft through the conceptual design phase . using raymer equations and CATIA software the final configuration and performance analysis has been achieved . the results have shown that the aircraft has a better performance, less fuel consumption and less maximum take-off weight than conventional aircraft.

5.2 Recommendation

- airfoil selection

the main problem that appeared during the work on this project is the selection of the airfoil , as the thickness of the chosen root airfoil was too small. It was hard to fit the number of passengers that was meant to be fitted in the wing-fuselage area, its recommended to design a special airfoil for this type of aircrafts taking all the criterias of design in consideration.

There are several methods for design and analysis of airfoil available for example:

- PROFIL
- XFOIL

- distributed proplusion

recent analytic and experimental distributed propulsion studies suggest several improvement in aircraft perform they include fuel consumption efficiency , noise abatement , steep climbing for short take-off & landing (STOL) . novel control approaches and high bypass ratio.it has also been suggested that smaller propulsors will be cheaper to manufacture and easier to handle during assembly and maintenance.

- stability check

the blended wing body is known to have a major stability problems , in both pitch and yaw motions , in order to maintain its stability in longitudinal and lateral axis an implementation of a vertical tail or winglets should be included.

As for the control of the aircraft , its preferable to combine both elevator and aileron into one control surface due to the absence of horizontal stabilizer . the rudder can also be implemented in either the vertical stabilizers or the winglets.

softwares recommended to analyze stability are , AVL (Athena vortice lattice method) to offer aerodynamic analysis & trim calculation and dynamic stability analysis .

stability and control digital DATCOM can be used to calculate the static stability , control and dynamic derivative characteristics of fixed wing A/C .

5.3 Future work

- for those who like to precede the preliminary phase , it includes the wind tunnel testing and computational fluid dynamic (CFD) calculations of the flow field around the A/C. major structural and control analysis is also carried out in this phase.
- for the detailed design phase you simply deal with the fabrication of the A/C to be manufactured , it determines the number , design and location of ribs , spars and other structural elements . flight simulators are developed at this stage .
- all of the obtained results (from calculations or drawings) can not be verified at this stage of design (conceptual phase) , because of the unavailability of equations , tools and sources. The only way it can be verified is through suitable wind tunnel to test its aerodynamic characteristics , performance and ground tests at latter stages of design to examine its stability .
- A stress analysis program such as NASTRAN PATRAN should be used to verify stress analysis on structure.

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Chapter three :

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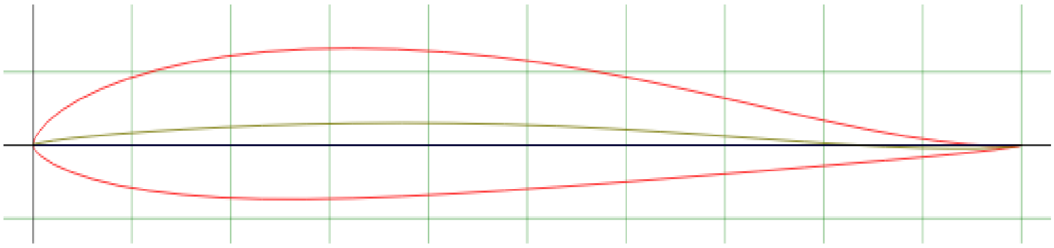
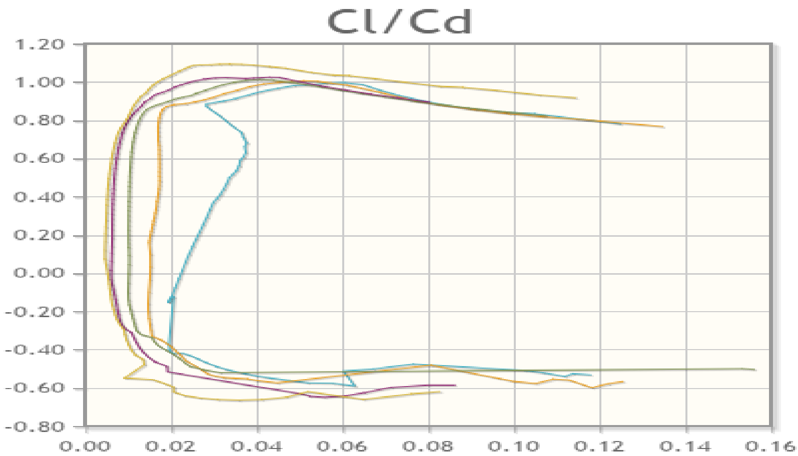
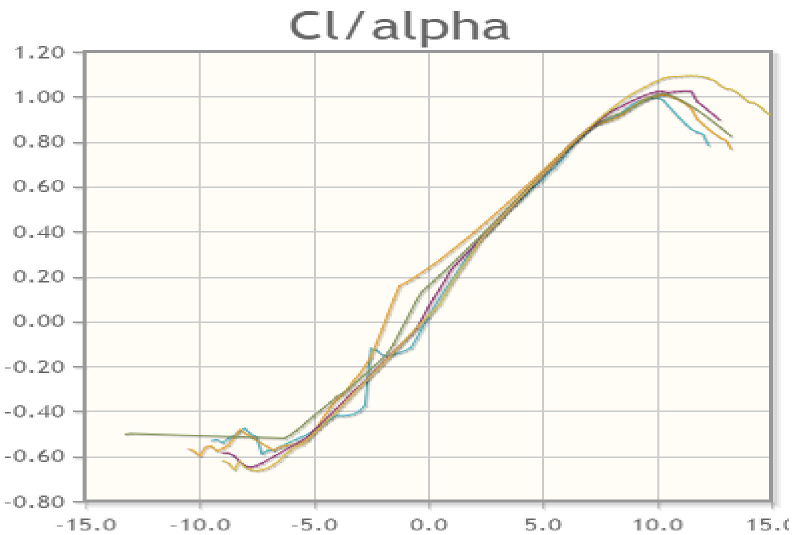
Appendices:

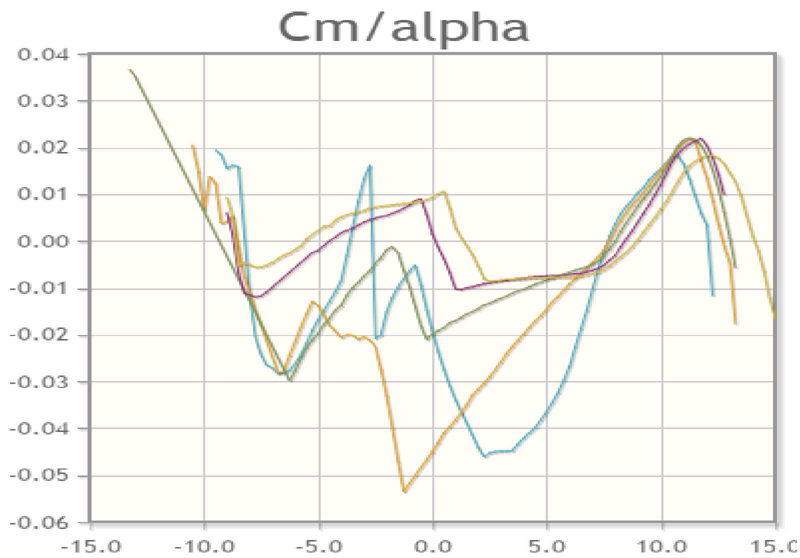
MH 61 10.26% (mh61-il)

MH 61 10.26% - Martin Hepperle MH 61 for flying wings





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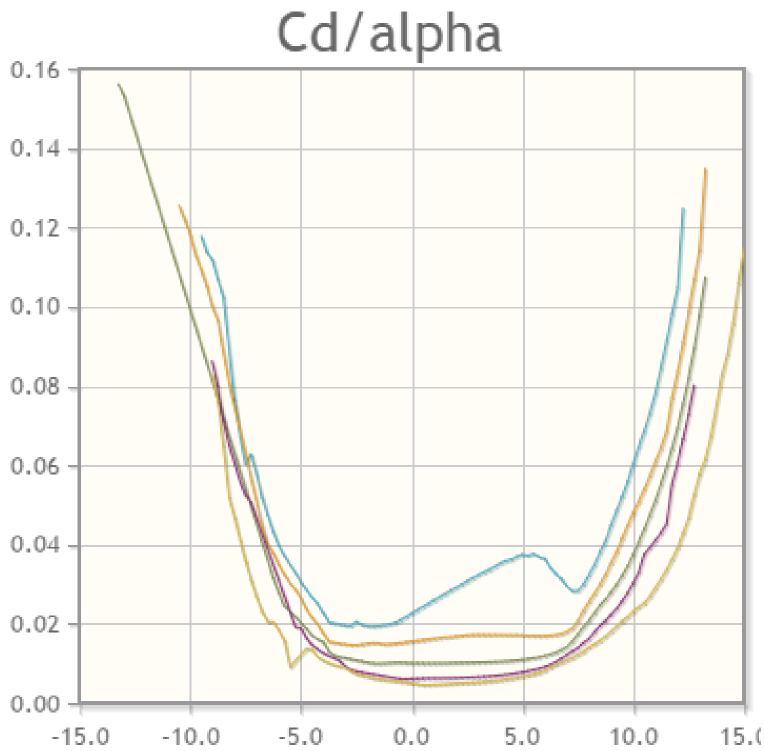
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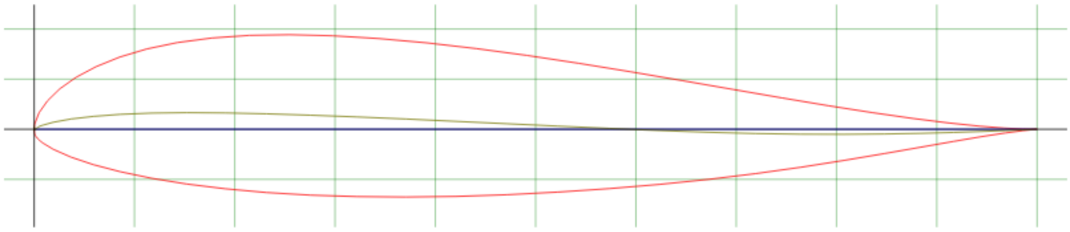
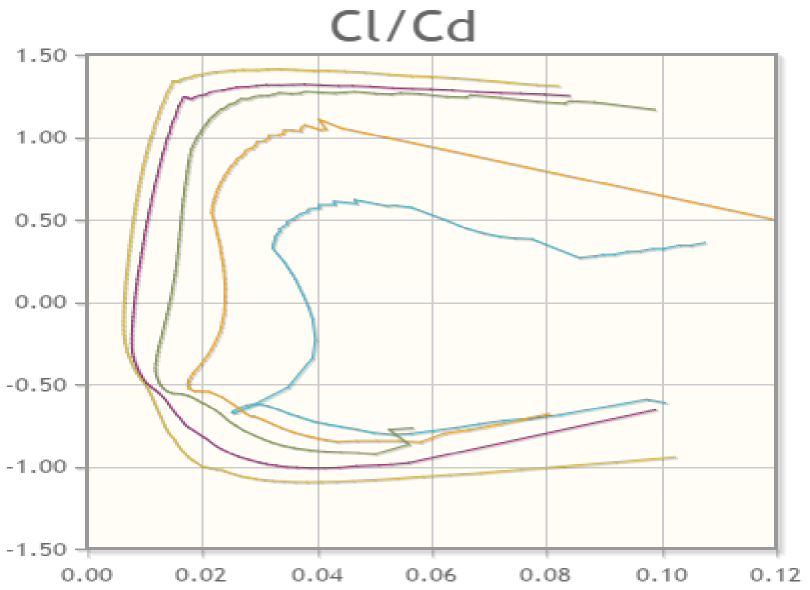
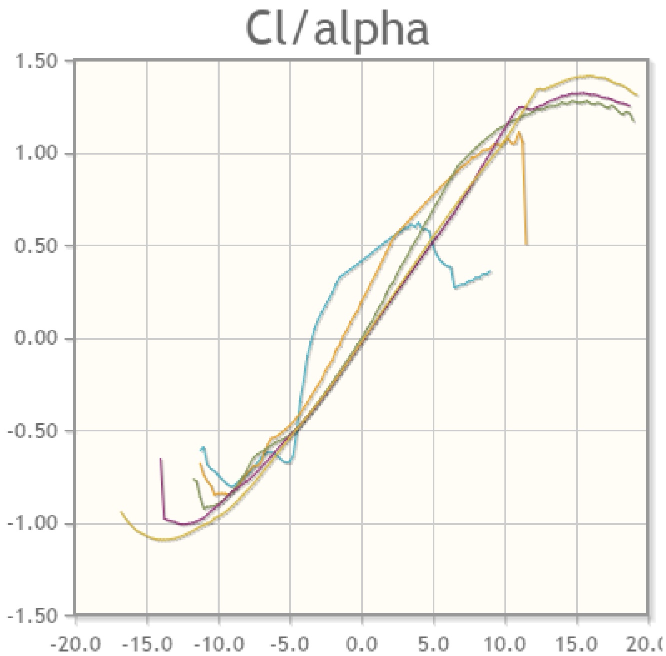
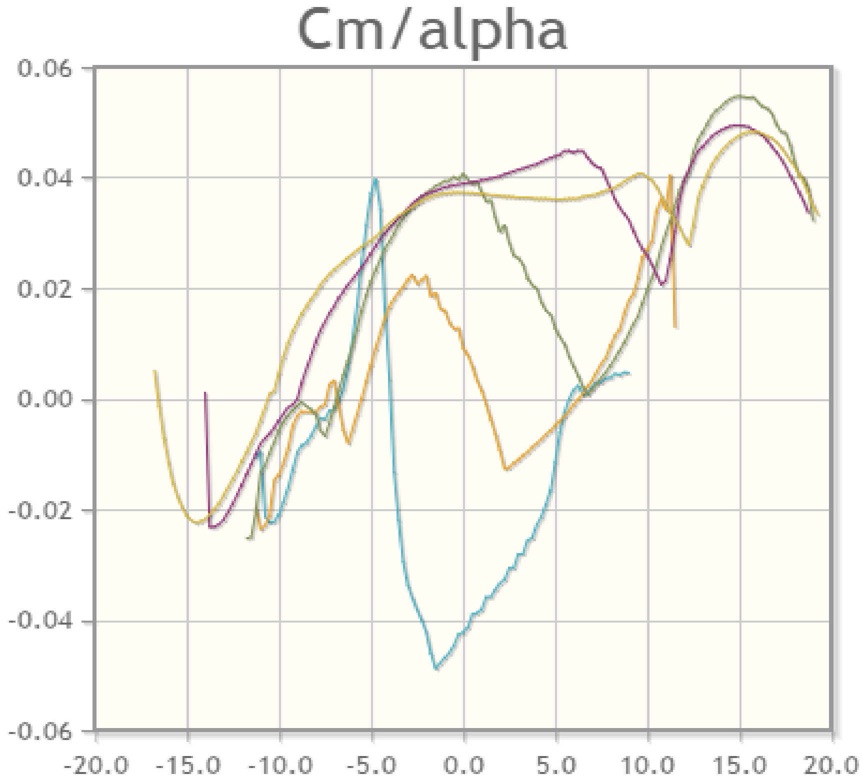
MH 93 15.98% (mh93-il)

MH 93 15.98% - Martin Hepperle MH 93 parafoil









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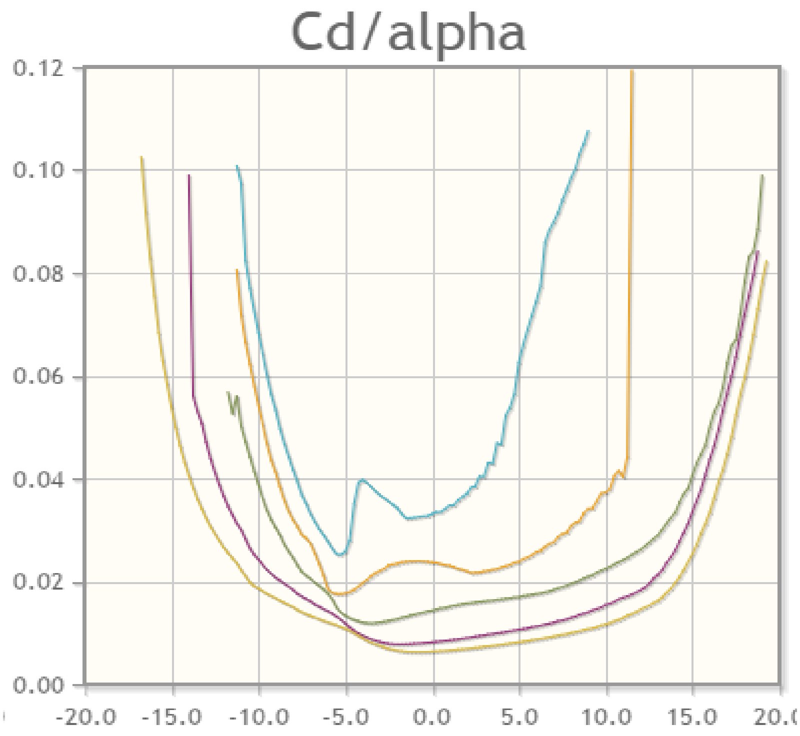

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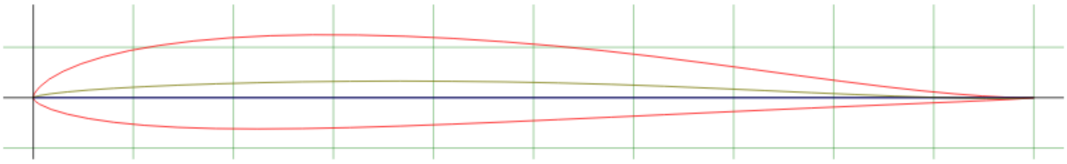
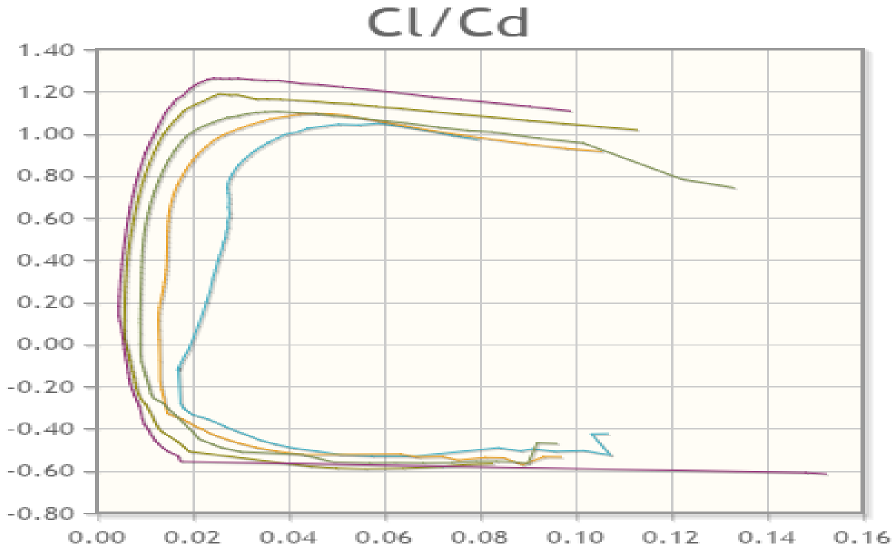
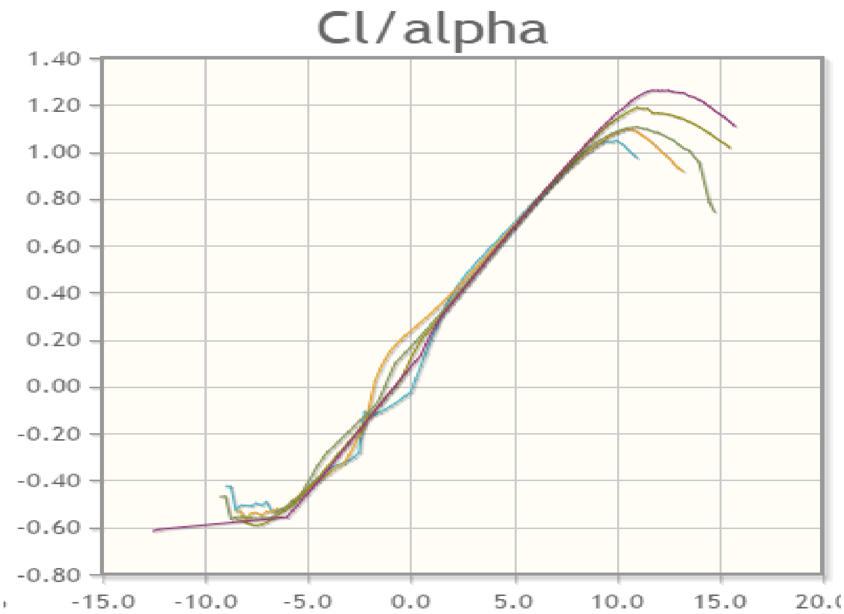
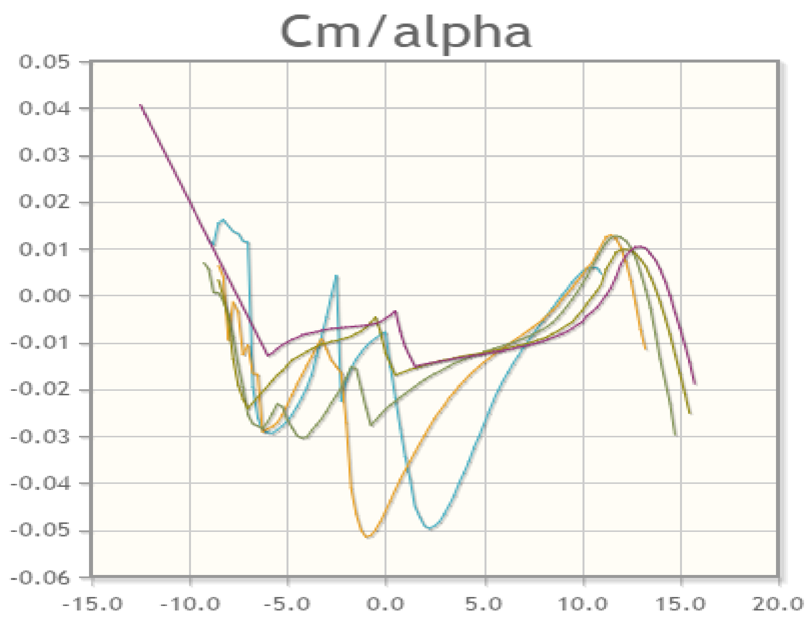
MH 62 9.3% (mh62-il)

MH 62 9.3% - Martin Hepperle MH 62 for flying wings





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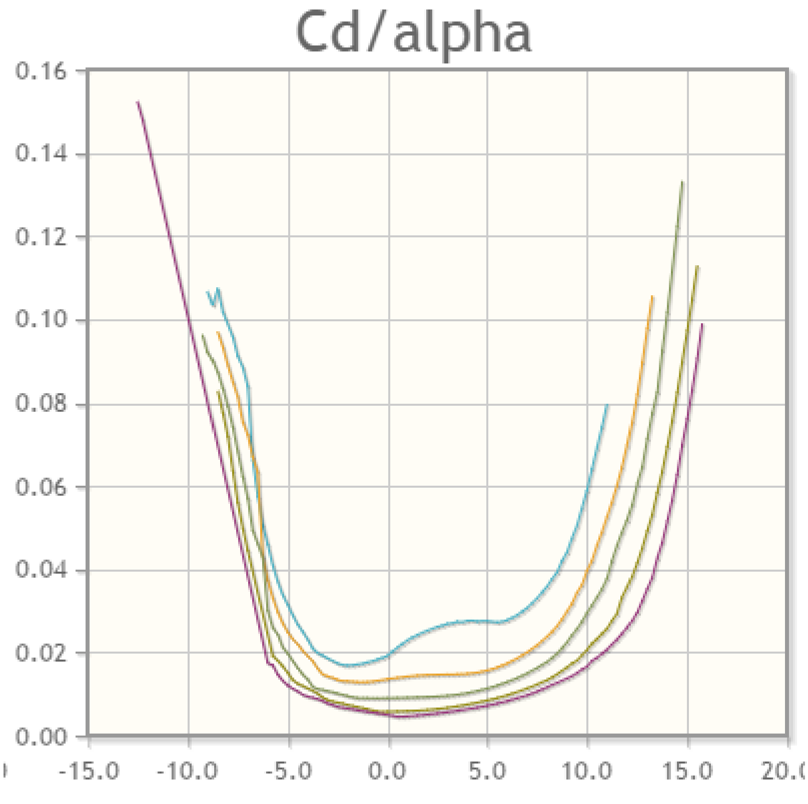
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