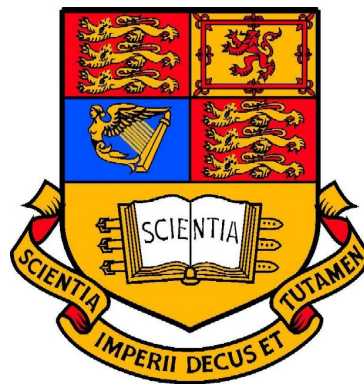


Design Synthesis of Advanced Technology, Flying Wing Seaplanes

by

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Declaration

I hereby certify that the research presented in this thesis has been carried out at Imperial College London, and has not been previously submitted to any other university for any degree or award. The thesis comprises only my original work. Due acknowledgments are made where appropriate.

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Abstract

Over the past decades there has been increasing pressure for ever more efficient and environmentally friendly aircraft to be designed. The use of waterborne aircraft could be a means of satisfying those requirements in the future. The aim of the PhD research program presented in this thesis was to develop the methodologies necessary for the preliminary design of large passenger seaplanes and evaluate the performance of such an aircraft compared to the current state of the art. The major technological and operational constraints in designing large waterborne aircraft were identified through an extensive feasibility study. A number of subject areas necessitating further investigation were also identified. To ensure that waterborne takeoff distance requirements are met, a novel initial sizing methodology was generated, relating the aircraft's thrust and lifting characteristics to the take-off Balanced Field Length. To allow the design of a broad family of aircraft based on a predefined baseline configuration, the seaplane geometry was fully parameterized. The aerodynamic properties of the entire aircraft were determined using a vortex-lattice potential flow solver, written specifically for the configuration being investigated, combined with other commonly used empirical methods. Novel methodologies for estimating the hydrodynamic characteristics of a broad range of parametric hulls were developed using the wealth of experimental hydrodynamic test data available. These methods can be used not only to predict the resistance and trim characteristics of a seaplane throughout the entire takeoff and landing manoeuvre but also give an initial estimate of the attitudes where hydrodynamic instabilities may be encountered. The airborne and waterborne performance characteristics of each resulting aircraft design were estimated using the aforementioned methods. The resulting design synthesis has been integrated into a single algorithm, written in FORTRAN, intended to allow the easy and prompt analysis of any parametric variant of the baseline configuration.

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Nomenclature

a	Speed of sound	m/s
A_c	Engine intake area	m^2
A_f	Engine fan area	m^2
b	Wing span	m
B	Hull beam	m
BFL	Balanced field length	m
BPR	Engine bypass ratio	
$\bar{c}$	Mean aerodynamic chord	m
C_D	Drag coefficient	
C_{D_o}	Zero-lift drag coefficient	
C_{D_i}	Induced drag coefficient	
C_{D_w}	Wave drag coefficient	
C_f	Friction coefficient	
C_M	Pitching moment coefficient	
C_L	Lift coefficient	
C_v	Velocity coefficient	
C_Δ	Hull beam loading	
e	Oswald efficiency	
F_∇	Displacement Froude number	
g	Gravitational acceleration	m/s^2
h	Altitude	m
h_1	Hull main step height	m
L	Aircraft centre section length	m
L/D	Lift to Drag ratio	
M	Mach number	

M_{cr}	Critical mach number	
M_{dd}	Drag divergence mach number	
N_{eng}	Number of engine	
N_e	Number of engines	
p	Roll rate about CG	$1/s$
q	Rate of pitch about CG	$1/s$
r	Yaw rate abt CG	$1/s$
Re	Reynolds number	
S_{ref}	Wing reference area	m^2
S_{TO}	Takeoff distance	m
S_L	Landing distance	m
sfc	Engine specific fuel consumption	kg/Ns
T	Thrust	N
V_∞	Freestream velocity	m/s
w	Density of water	kg/m^3
W_e	Aircraft empty operational weight	N
W_f	Fuel weight	N
W_o	Maximum takeoff weight	N
W_{old}	Payload weight	N
x_{np}	Longitudinal aircraft neutral point	m
$\mathcal{R}$	Reference wing aspect ratio	
α	Angle of attack	deg.
β	Sideslip angle	deg.
β_1	Hull deadrise angle at main step	deg.
Δ	Weight supported by hydrodynamic & hydrostatic forces	N
δ_k	Afterbody keel angle	deg.
δ_s	Hull step angle	deg.
λ	Reference wing taper ratio	
$\Lambda_{c/4}$	Quarter chord sweep angle	deg.
τ	Hull trim angle	deg.
τ_o	Forebody keel angle	deg.
ρ	Density of air	kg/m^3

Chapter 1

Introduction

1.1 Motivation

The commercial aviation industry is currently experiencing an unprecedented boom, with both passenger and cargo traffic expected to treble by 2025 [10, 28]. This continued growth comes at a time when concerns about the environmental impact of aviation are on the rise.

Noise pollution caused by low flying aircraft remains a major concern for areas surrounding airports, despite a considerable improvements over the past 40 years. However, the projected increase in airport traffic over the next two decades is very likely to counterbalance any (limited) further reduction in aircraft noise [22].

The environmental effects of aircraft also have a knock-on effect on airports and airlines. Over the past decade there has been a substantial increase in the number of airports with noise-related restrictions; including noise limits, curfews, quotas, and fines [11]. Concerns over increased noise pollution and further deterioration in the local air quality are also starting to hinder airport expansion plans worldwide. This at a time, when both aircraft noise and runway capacity have been recognised as significant sources of airport congestion constraints for Europe and North America [6].

Currently 93 major airports, that handle 63% of the worldwide traffic, are capacity constrained [6]. A further increase in traffic, without a comparable increase in airport capacity, would lead to more delays and increased competition for landing slots, ultimately increasing airline operating costs.

Large modern seaplanes could be used to address these problems. The adoption of a blended wing body/flying wing design was proposed by Serghides [81] as a possible solution to some longstanding seaplane performance issues. Moreover, the lack of fixed runway and taxiway dimensions and the

adoption of a versatile design for the apron areas, would allow aircraft to be optimised for their intended mission profile without having constraints set on their dimensions. Aircraft size, payload and range could therefore be as large as necessary for maximum efficiency or profitability. Furthermore, the majority of takeoff and landing paths can be placed entirely over water, moving low-flying aircraft away from densely populated areas.

Through use of the Blended-Wing-Body (BWB) configuration and modern technology, future seaplanes would not suffer from the reduced efficiency that plagued seaplanes in the past. The BWB, with its superior aerodynamic design and using highly efficient high-bypass ratio engines mounted on top of the fuselage, has demonstrated improved fuel efficiency, reduced emissions and a significantly lower noise signature compared to the conventional tube-and-wing aircraft. In fact, the lack of constraints on the aircraft's span should have a favourable effect on the aircraft's Lift-to-Drag ratio as, according to Green [32], it is proportional to a ratio of the wing span to a function of the wing's wetted area.

1.2 Objectives

The aim of the project presented in this thesis is the development of a complete initial and preliminary design methodology for waterborne aircraft of blended-wing-body or flying-wing design. Given the novelty of this concept, new methodologies had to be generated where necessary. Particular attention must be given to the performance of the aircraft on water, as the waterborne takeoff thrust requirement is a major constraint in designing seaplanes. Several methodologies of varying degrees of complexity are therefore created and applied at different stages of the synthesis process. Furthermore the aerodynamic effects of the seaplane hull must be quantified and the advantages of using a BWB configuration identified. These methods must subsequently be incorporated into a single computerised aircraft synthesis that can provide a flexible, accurate way of evaluating the performance of the seaplanes in question. Consequently, by evaluating the performance characteristics of several sample aircraft, the feasibility of resurrecting the seaplane can be determined and best practice rules can be established for future designs.

1.3 Background

1.3.1 Seaplanes

Seaplanes are aircraft capable of operating from large bodies of water such as the sea, rivers and lakes. Seaplanes fall into two main categories: flying boats and floatplanes. The fuselage (hull) of a

flying boat serves a dual purpose in providing both volume for payload and buoyancy for the aircraft. On the other hand, a floatplane's fuselage is clear of the water at all time and slender floats (pontoons) are mounted underneath the hull to provide buoyancy.



(a)



(b)

Figure 1.1: Members of the floatplane family: (a) the deHavilland Canada DHC-6 "Twin Otter"; (b) the Douglas XC-47C



(a)



(b)

Figure 1.2: Members of the Flying Boat family: (a) the SaRo SR-45 "Princess" Flying Boat; (b) the Beriev Be-200 Amphibious Jet

Floats are mostly found on smaller aircraft, such as the DHC-6 "Twin Otter" (fig. 1.1(a)), and are occasionally used to modify existing landplanes for waterborne operations, such as the Spitfire Mk.IX and the XC-47C, a modified Douglas DC-3 (fig. 1.1(b)). They are most often seen in a twin side-by-side configuration, however there were cases where single or three floats were used. Although allowing aircraft to operate from water with limited structural modifications, the use of floats leads to large weight and drag penalties. Floats have also been found to adversely affect the handling performance of an aircraft.

The overall configuration of a flying-boat closely resembles that of most conventional tube-and-

wing landplanes. However several modifications are required to facilitate waterborne operation. The underside of the fuselage is V-shaped to reduce the landing impact loads and steps are used to reduce hydrodynamic suction forces during takeoff. To keep the lifting surface and propulsive systems away from water and spray, wings are traditionally mounted high on the fuselage with the engines mounted on top of the wing. Smaller pontoons are often placed on the tip of the wings to give the aircraft additional rolling stability when in the water. The larger size of flying boats allows them to operate in much higher sea states than floatplanes, while their "cleaner" configuration leads to a more efficient aircraft.

A subcategory of both floatplanes and flying-boats are amphibious aircraft. Through the use of a retractable undercarriage system, landing on paved runways or beaching under their own power is possible. Their amphibious nature allows them to engage in a larger array of mission profiles and to be based away from water, thus reducing maintenance costs. However the addition of an undercarriage system results in considerable weight penalties, making amphibians unattractive for missions that can be fulfilled by landplanes.

A Brief Historical Overview

The first airworthy seaplane to take to the skies was the French "Le Canard" on March 28th 1910. Over the following years, advances in propulsion and seaplane design lead to the first transatlantic flight of a US Navy, Curtiss NC-4 Flying Boat. The lack of readily available runways, the increased safety of being able to land on water and the large capacity of flying boats resulted in their adoption for long range flight by the Imperial Airways and Pan-American Airways.

The National Advisory Committee for Aeronautics (NACA) continued research into seaplane hydrodynamics and design until the early 1950's. However the great improvements in landplane performance witnessed during World War II and the ready availability of paved runways worldwide lead to the triumph of the landplane over the waterborne aircraft. In the late 1950's, some attempts at using jet power and bringing the seaplane back into the foreground were made. In the UK, Saunders-Roe continued seaplane development with the 100 seat Saunders-Roe SR-45 "Princess" (fig. 1.2(a)), the jet powered Duchess and the SR.A/1 fighter. A role for the seaplane within the US Navy was also envisioned, giving rise to the Martin P-6M "SeaMaster" jet bomber, the Convair R3Y-1 "Tradewind" troop transport and aerial refueler and the Convair F2Y "Sea Dart" supersonic fighter. All these development programs showed enormous potential, however numerous design flaws and the political and economic circumstances of the period lead to their cancellation.

With many of their original roles, such as Search and Rescue (SAR) and Anti-Submarine Warfare,

being performed by landplanes or helicopters, limited roles remained for seaplanes in the west. In Japan the Shin Meiwa US-1 has performed brilliantly in SAR operations, giving its operators increased cargo capacity, speed and range. Further, amphibians such as Canadair's CL-215, its successor the CL-415 and Beriev's Be-200 (fig. 1.2(b)) are often the only means of effectively fighting large forest fires. In the civil aviation sector, floatplanes and small amphibians continue to be used for recreational purposes and small scale passenger operations in regions where landing strips are not readily available (bush flying).

1.3.2 The Flying Wing

The concept of a flying wing is not a novel one either. It has long been identified by aerodynamicists as the optimum configuration for reducing drag and maximizing volume for a given wingspan.

The Northrop Corporation attempted to introduce the flying wing into USAF service as a bomber in two occasions. After the proposal for the YB-49 fell through, J. K. Northrop proposed converting the bomber into an 80-seat passenger transport. However, the lack of any surfaces other than the wing lead to a number of stability issues which could not be rectified until the introduction of modern fly-by-wire control systems. Through use of such advanced systems the Northrop B-2 Spirit became the first large flying wing to successfully enter service.

Recently, following concerns over the environmental impact of conventional aircraft, there has been an increased interest in large flying wing transport aircraft. Boeing's X-48B Blended Wing Body (BWB) concept, once rumoured to be the company's 797 aircraft, is at the forefront of this drive. Further, teams from Cambridge University and the Massachusetts Institute of Technology have been collaborating on the design of the SAX-40, in an attempt to minimise an aircraft's noise signature and fuel consumption. Numerous additional concepts have been presented by universities and aerospace research organisations such as TsAGI, NASA, DLR and ONERA.

Chapter 2

Feasibility Study

Given the lack of any significant research in seaplane design over the past 50 years, it is imperative that a feasibility study is conducted. Examining both the current and possible future requirements of civil and military aviation, possible applications, operational scenarios and design characteristics for a new generation of flying boats can be identified.

2.1 Possible Uses of Large Seaplanes

2.1.1 Passenger and Cargo Transport

As stated in section 1.1, the use of waterborne aircraft could mitigate certain environmental concerns associated with the rapid growth of civil aviation. Using seaplanes can drastically reduce the number of populated areas overflown at low level and thus the associated noise; by relaxing aircraft size constraints, given the lack of paved runways, taxiways and apron areas, larger and more efficient aircraft could be designed.

From an operational standpoint, the most important requirement for the use of ultra-high capacity seaplanes as long-haul passenger transports is the proximity of major hubs to the sea, as this would facilitate the transfer of passengers and cargo from seaplanes to landplanes or amphibians. Currently most large airlines use hub airports as the centre for their operations, offering connecting flights to and from other airports (spokes). It is estimated that approximately 77% of the world's long haul traffic originates from hubs in 32 major world cities [6] and that traffic is likely to increase as these megacities grow larger and wealthier. As seen in figure 2.1, twenty-four of these cities are within 50km of a large body of water. In fact, sixteen of these cities have a major international airport bordering the water, while another three cities have smaller (single or twin runway) coastal airports.

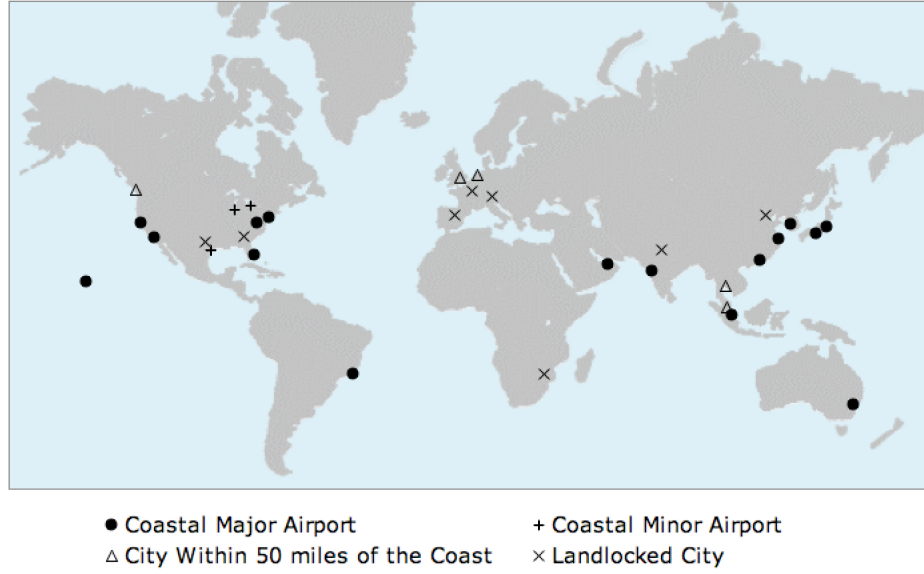


Figure 2.1: The proximity of 32 major hub cities to water

Air freight carriers are likely to find the increased cargo capacity attainable by future waterborne aircraft very attractive, particularly given the growth forecasts for the sector. Although, cargo carriers use regional hub airports through which most cargo is routed, these hubs do not have to be situated near major cities, thus facilitating the establishment of a seaplane hub near an existing smaller airport.

2.1.2 Maritime Search and Rescue

An ideal vehicle for maritime Search and Rescue (SAR) should be able to fly to the scene of a distant accident, loiter over the area to locate survivors, retrieve them, provide first aid and promptly transport them to dry land. Even though helicopters suffer from limited payload capacity, range speed and endurance, improvements in their performance and their ability to hover over an area have made them the vehicle of choice for SAR. On the other hand, seaplanes incorporate all the requirements for an effective SAR aircraft as repeatedly demonstrated by the Japanese ShinMaywa US-1A. The US-1A can cruise at 230 knots for 2300 nautical miles and rescue up to 12 people in sea-state 4 waters [92]. This amounts to approximately double the speed, three times the range and a higher passenger capacity than most current SAR helicopters. A review of US Coast Guard rescue records from 1993 to 1995 revealed that an additional 7 to 11 lives could have been saved if seaplanes were used for rescue operations at a distance of 100 km or greater from the US coast [14].

2.1.3 Firefighting

Aerial firefighting is a critical tool in combating wildfires. This is one of the only missions for which seaplanes are still used, as they can scoop up water by landing on lakes or the sea. In water bombing the larger the amount of water dropped, the better the result. Therefore a larger seaplane would be able to combat the flames more effectively. However two other critical factors are the maneuverability and climb rate of the aircraft, as water bombers often have to fly at low levels over mountainous terrain. Meeting these requirements would probably lead to the use of additional control surfaces, such a canard or a V-tail, a modification that would certainly increase the aircraft's drag.

2.1.4 Troop and Cargo Transport

Rapid deployment of troops and equipment to any battle theater is a major requirement for any major modern military. Attempts at using ultra-high capacity seaplanes for strategic airlift have experienced varying levels of success. The H-4 Hercules, better known as the "Spruce Goose" due to its all wooden construction, was commissioned during the 2nd World War to transport up to 750 fully equipped troops. Another example of ultra-high capacity waterborne vehicles used for strategic lift are the Soviet KM and Lun class Ekranoplans and Boeing's Pelican WIG concept.

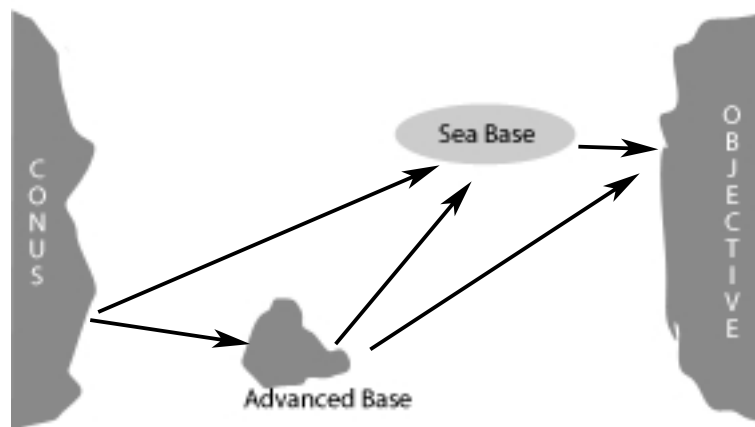


Figure 2.2: Diagram of the US Navy's SeaBasing Concept

Seaplanes are now given another chance with the adoption of the SEA BASE concept (Fig. 2.2) by the US military [24]. The need for rapid strategic lift between the mainland, advanced support bases and the sea base has lead to a number of studies by the US Navy into seaplanes.

Heavy lift need not be the only application for transport seaplanes within the military. Smaller amphibious aircraft could be used to rapidly land troops and vehicles during an amphibious assault,

much like the Convair P3Y-2 Tradewind was intended to. Recently, the U.S. Defense Advanced Research Projects Agency (DARPA) has revisited the idea of deploying small Special Operations teams using a waterborne aircraft, with a airborne range of 1,000nm and a waterborne range of 200nm. In this case however, the craft would also have to be submerged for the last 11nm of its journey [23].

Further uses could include rapid support for submarines and deployment and retrieval of Unmanned Undersea Vehicles (UUV) or Unmanned Surface Vehicles (USV).

2.1.5 Electronic and Anti-Submarine Warfare

The large cargo capacity of the seaplane under investigation, would make it the ideal aircraft to be outfitted with a wide range of jamming, surveillance and communications equipment. Possible missions would include: Airborne Communications, Command & Control (C³), Electronic Warfare and Maritime Surveillance. The ability of seaplanes to land and loiter on water could prove quite beneficial in Anti-Submarine warfare missions and increase the time on station when on a C³ mission.

2.1.6 Aerial Refueling

The potential use of BWBs as aerial refuellers is already being investigated by Boeing through their X-48B program. The lack of size constraints in seaplane design would allow sea based aerial refuelling aircraft to carry more fuel, improving their endurance or increasing the number of aircraft that could be refuelled. Being waterborne would also allow these aircraft to land in the open ocean while waiting for an assignment, saving on fuel and allowing them to remain on station longer.

2.2 Technological Feasibility

Despite the lack of any substantial research into seaplane design or the development of seaplane specific technologies, a modern BWB seaplane could greatly benefit from the advances seen in aeronautical and marine engineering over the past fifty years. The reasoning behind choosing a BWB configuration is presented in section 4.1.1.

2.2.1 Fluid Dynamics

2.2.1.1 Wing Design

The advantages of BWB aircraft over conventional configurations have lead to a number of research programs over the past decade. However due to the relative novelty of this concept there is

no established best practice for BWB design. Given the transonic nature and complex design of flying wings, computer based solutions, such as panel codes or Computational Fluid Dynamic (CFD) modelling, have been extensively used when designing and optimising airfoils and wing planforms. Overall, given the extensive amount of information on BWB design methodologies, the aerodynamic design and optimisation of the wing planform and airfoils should not present any problems. It should however be noted that, given the issues associated with taking off and landing on water, the aircraft's stall speed will have to be minimised. This can be achieved using high lift devices such as slats and lower wing loadings, while the possibility of using trailing edge flaps should be investigated.

2.2.1.2 Hull Design

Flying boats commonly use their hull to land on water. The need to compromise some of the hull's aerodynamic efficiency, in order for hydrodynamic and seakeeping requirements to be met, is perhaps the biggest drawback of seaplanes. Occasionally retractable hydroski, hydrofoil or air-cushion systems have been employed to reduce water resistance and landing impact loads while maintaining a cleaner aerodynamic hull shape. The merits and disadvantages of these hydrodynamic configurations should be reviewed.

Planing Hull

Most flying boats, such as the one seen in figure 1.2(a), can be described as monohull planing vessels; as hydrodynamic pressure loads support most of its weight, with buoyancy adding but little support. The hydrodynamic and aerodynamic pressure loads, lift a great portion of the boat out of the water, reducing the area wetted by the denser fluid and thus its overall resistance.

A major drawback of using the hull as a planing surface, is the need to introduce a step to counter the hydrodynamic suction forces generated by the curvature of the afterbody as a result of the Coanda Effect. While in flight, the step causes the airflow to separate and causes vortical structures to form, adding as much as a 38% drag penalty . Using a retractable fairing for the step, its adverse aerodynamic effects could be minimised. Further, alternate hull shapes stemming from advances in speedboat, hydroplane and WIG design could be utilised.

The bottom of the hull would also have to be V-shaped in order for impact loads during landing to be reduced. Given the relatively wide centrebody of BWB aircraft, such shaping would lead to an increase in wetted area and a relatively low length-to-beam ratio. A decrease in wetted area could be obtained by employing a multihull configuration. In a catamaran configuration, the spacing between

the two slender hulls would also give the aircraft additional stability while on the water and interference between the hulls could reduce the resistance caused by wave generation. Further, the use of hulls with higher length-to-beam ratios has been found to reduce resistance and spray generation, as well as improve rough water performance. Blending two smaller V-shaped hulls into the fuselage would also be easier, as the body's airfoil shape would not be affected throughout.

Hydrofoil

The idea of using a retractable hydrofoil to reduce resistance during takeoff has been attempted several times. A hydrofoil is a submerged wing that provides lift, raising the vessel out of the water and therefore greatly reducing its resistance. Another advantage of hydrofoils is that they show better seakeeping characteristics than semi-displacement or planing vessels, as they are not as strongly affected by sea waves. Operating in water allows a wing to generate forces one thousand times higher than in air, however when speeds in excess of 50 knots are reached the foil tends to cavitate, leading to a sudden loss of lift. Supercavitating airfoils could be used to overcome this problem however they do not perform as well as subcavitating ones.

Due to the loadings it would be subjected to, the use of a hydrofoil during landing is very unlikely. Therefore despite the great weight penalty of installing such a system, the fuselage would still have to be shaped for marine operations and strengthened to withstand the landing impacts, reducing the potential for reduction in aerodynamic drag and structural weight.

Hydroski

A hydroski is effectively a flat plate mounted on the bottom of the hull. It was used on the Corvair Sea Dart to allow its clean slender hull to takeoff and land from water. Hydroskis have also been used to reduce the landing impacts of seaplanes and to dampen wave induced motions [88]. A hydroski could be designed much like a retractable undercarriage, allowing for the minimum amount of alterations to the aircrafts underbelly, thus minimising the resulting aerodynamic and weight penalty. However, the hydroski has been found to produce much hydrodynamic resistance before it planes, making it unsuitable for aircraft with little excess thrust.

2.2.1.3 Extreme Ground Effect

One of the major requirements in seaplane design is the reduction of the takeoff speed, as it limits the thrust required for takeoff. Using Extreme Ground Effect (EGE), the aircraft could lift off at lower speeds and accelerate to its takeoff speed while airborne. EGE, also known as chord-dominated ground

effect, is experienced by airfoils or wings flying at a height much smaller than their chord length. The low ground clearance forces the flow under the airfoil into becoming channel flow, thus affecting the flow field around the wing and increasing its lift coefficient. EGE has been used extensively by WIG craft but not large flying boats, as their wings are mostly mounted high to avoid water damage and spray ingestion by the engines.

Instead, the flying wing configuration allows for use of ground effect; the centrebody being a thick airfoil of very large chord length that lies very close to the sea. The use of a catamaran configuration may further increase the lift coefficient, as the twin hulls would act as end plates.

However, certain issues have to be addressed regardless of whether the centrebody is optimised to utilise extreme ground effect or not. Particular concern arises from the movement of the airfoil's aerodynamic centre from the quarter-chord point to the one-third chord point, due to the altered flow field around the airfoil. This movement, coupled with the increased lift coefficient, leads to very large nose-down pitching moments which can affect the vehicle's stability. S-shaped airfoil sections have been used in the past to reduce these pitching moments, but they have demonstrated very poor performance characteristics out of ground effect. Therefore, further research into airfoil designs that perform well both in and out of ground effect is required.

2.2.2 Propulsion

The adoption of modern High Bypass Ratio (BPR) turbofan engines has resulted in reductions of greenhouse gas emissions, noise and specific fuel consumption (sfc), the measure of fuel required per unit of thrust. The exact position of the engines on the aircraft will be determined by airworthiness and seaworthiness requirements. On existing BWB designs, the engines are placed toward the trailing edge of the centrebody. By that point the boundary layer is quite thick, traditionally requiring the engines to be mounted on pylons.

Another option is the use of a boundary layer ingesting propulsion system. These engines are blended into the fuselage, ingesting the boundary layer and reenergising the aircraft's wake, in order to reduce drag. The use of emerging propulsion systems such as the geared turbofan and Ultra High BPR engines, known as propfans, should also be explored. Should such novel propulsion systems be used, the possibility of spray damaging the fan blades and the possibility of using thrust reversal should be investigated.

Although water ingestion is not likely to be a problem if the engines are mounted on struts, blending them into the fuselage would bring them substantially closer to the water. Due to this proximity, the likelihood of water entering the engines when the aircraft is moored or loitering in tumultuous

water should be investigated and possible solutions should be found.

The thrust-to-weight ratio required for a seaplane should also be investigated. The minimum thrust requirement is determined by the drag experienced during cruise or the maximum resistance encountered during takeoff, however having a moderate amount of excess thrust is advisable for seaplanes. A survey of the power-to-weight and thrust-to-weight ratios of existing seaplanes should be done. Relating the thrust to weight ratio to takeoff distance should also be considered, as such a relation would be necessary in the conceptual design stage. It is expected that the thrust-to-weight ratio for a seaplane BWB is going to be larger than that of an equivalent landplane.

2.2.3 Stability and Control

2.2.3.1 Airborne

The delayed adoption of the flying wing aircraft configuration can be partly attributed to our inability to effectively stabilize and control them in the past. Since there have been massive advances in electronics, control systems engineering and vehicle design. Now, not only are BWB aircraft considered a viable option for mass air travel, but fly-by-wire control systems are robust enough to be adopted by aircraft such as the Airbus A380 and Boeing's 787. Therefore the stability problems encountered in the past can be overcome through appropriate design of the wing, placement of the CG and use of modern control systems.

As stated in section 2.2.1.3, stability problems may be encountered during takeoff and landing as a result of increased pitching moments generated by wing sections when in EGE. The extent of this problem for a BWB seaplane is unknown at this stage, as only the centrebody section rather than the entire wing will experience the effects of EGE. In WIG craft, where the problem is quite pronounced, it is controlled by increasing the size of the horizontal stabilizer. As BWB aircraft have no such surfaces, other methods have to be employed. Use of certain airfoils sections and wing planforms, CG control and the use of control surfaces are all likely solutions. Some consideration should also be given to the peculiarities aircraft stability in EGE, detailed by Synitsin [91].

The control layout of the BWB seaplane should not differ much from that of the equivalent landplane. Pitch and roll control shall be provided by hinged elevons distributed along the trailing edge of the aircraft. Winglets, which are used to reduce lift-induced drag and augment lateral stability, can

be fitted with rudders to provide yaw control. Should additional yaw control be required, outboard elevons could be modified to also become drag-rudders. In order for landing distance and bouncing to be minimized, spoilers will be fitted along the outboard wings to reduce lift and increase drag.

2.2.3.2 Waterborne

Longitudinal and lateral stability will not be an issue when a BWB seaplane is moored or moving at low speed. The vessel is expected to be statically stable, as the large length and width of its waterplane area will place the metacentre higher than its center of gravity. The aircraft being statically stable in the lateral direction would also negate the need for tip-floats to be used. In conventional aircraft these floats serve to produce a righting moment when the aircraft rolls on water. In flight however, they generate substantial amounts of aerodynamic drag. Aerodynamic drag can be reduced using retractable floats, however the weight penalty will remain or worsen.

The main problem will arise in accurately predicting stability when the seaplane is in the planing regime, something traditionally done through model testing. Static stability in the planing regime could be analyzed by considering the combined hydrodynamic and aerodynamic forcings the aircraft is subjected to. Several empirical and theoretical models, as well as published experimental results can be used.

FAR 25.231 to 25.239 specify the requirements for waterborne handling characteristics. While on the water the pilot will require adequate yaw control to steer the seaplane in any direction. Using the seaplane's rudders alone is not possible as they do not produce sufficient forces to perturb the aircraft. One possible solution is deflecting all the control surfaces on one side of the aircraft to generate drag and thus a yawing moment. The aircraft's engines could be placed further outboard on the wings, to allow for yaw moment to be generated using differential thrust.

Avoiding the use of mechanical systems within the water is preferred from a maintenance perspective. However, should the yaw moments generated through aerodynamic and propulsive means be deemed insufficient, the use of retractable rudders is an option. Further, use of hydrodynamic drag generators, similar to an aircraft's air brake, placed underneath the hull could result in larger yawing moments at a lower weight penalty. Such "flaps" could provide further hydrodynamic pitch control, particularly at lower speeds.

2.2.3.3 Rough Water Performance

The ability of an aircraft to operate from water is limited by the local wave conditions. A sea-state number is often used to classify the sea in terms of significant wave height, the average height of the one-third largest waves encountered, as seen in table 2.1. The methods used to predict the likelihood of encountering a given sea condition can be found in section 2.3.1.1. It is estimated that the operations of most conventional seaplanes are limited to a sea-state 3 [88, 92], while more advanced seaplanes such as the Beriev A-40, ShinMeiwa US-1A and Corvair R3Y Tradewind can operate in wave heights as high as 2.2m (sea state 4).

Sea State	Description	Significant Wave Height (m)	Sustained Wind Speed (knots)
0-1	Calm	0-0.1	0-6
2	Smooth	0.1-0.5	7-10
3	Slight	0.5-1.25	11-16
4	Moderate	1.25-2.5	17-21
5	Rough	2.5-4.0	22-27
6	Very Rough	4.0-6.0	28-47
7	High	6.0-9.0	48-55
8	Very High	9.0-14.0	56-63
>8	Phenomenal	>14.0	>63

Table 2.1: The Sea State Code [29, 88]

Sea swells have multiple effects on aircraft operations. They affect the structural weight of the seaplane, as higher loads are experienced and certain hydroelastic effects have to be considered. During takeoff and landing, waves have been found to destabilize the aircraft, causing otherwise stable hulls to porpoise. Use of forebody warping and high length-to-beam ratio hulls has been found to reduce impact loads, widen the range of trims where porpoising is not encountered and reduce the tendency of the forebody to dig into the water. Use of higher length-to-beam ratios has also been found to made takeoffs less violent [15, 16].

Operation in waves also leads to a significant increase in resistance. For instance, planing in seas where the significant wave height is approximately 40% of the beam can increase the resistance by up to 15% compared to that in calm water [3].

In rough seas, the distance between consecutive swells (wavelength) is another important factor. The ratio between the height (amplitude) and wavelength of waves is an important factor. The higher it is, the likelihood of a vessel capsizing and the magnitudes of wave impact loads increase. Further, loitering on rough water of wavelength less than the aircraft's submerged length will impart longitudinal bending loads on the aircraft's fuselage.

The habitability of the seaplane when loitering or drifting on rough seas is also an issue. The International Organization for Standardization [41] defines the habitability standards, in terms of acceptable duration and heave acceleration amplitudes. The application of these regulations should extend to the motions experienced during takeoff and landing.

2.2.4 Configuration Layout

A BWB seaplane is likely to be configured much like equivalent landplane designs. The exact wing planform will depend on the aerodynamic requirements for transonic and EGE flight. Dihedral will be used to lift the outboard wing away from the water. The aircraft's underbelly will be configured so that fluid-dynamic drag is minimized and seakeeping performance is improved.

Internally, the payload will be housed in the centrebody section, with cargo compartment areas on either side and fuel tanks in the outboard wing sections. For passenger operations, certain complications arise due to the width of the cabin. When performing a turn, passengers further away from the aircraft's centerline will feel much higher accelerations than in current commercial aircraft. Furthermore, the aircraft's design limits the number of windows available to passengers. The use of screens displaying the external view has been suggested but it is rumored that it was not received well by passengers. Passenger evacuation is a major issue, as no satisfactory plan for emergency egress has been suggested and seaborne operations are only likely to complicate the evacuation requirements. The use of additional escape hatches on the top of the aircraft should be considered, as waterborne operations prohibit the placement of emergency exits on the aircraft's underbelly.

2.2.5 Structural Design

The centrebody of the aircraft is the crucial structural assembly. As part of the wing it is subjected to bending and torsional loads generated by the outboard wing section, as the passenger cabin it is subjected to pressurization loads and as the planing hull it has to withstand various hydrodynamic loads, specified in FAR 25.223 through 25.237. Unlike tubular airframes, the BWB's fuselage is not

optimally shaped to withstand pressurization loads. Furthermore, given the repeated loading and unloading cycles, resistance to fatigue damage will be critical.

The structural design could resemble that considered by Boeing for the X-48B [49]. Ribs are placed along the cabin, splinting it into multiple long passenger cabins, to preserve the aerodynamic shape. The wing and pressurization bending loads are carried by thick sandwich structures placed above and below the cabin. Suitable reinforcement of the lower structure could allow it to withstand the hydrodynamic impact load experienced during landings. However, the requirement by FAR 25.755 for watertight compartments between the external hull and the bottom of the passenger cabin is likely to complicate the structural design of the the fuselage.

A major constraint in materials selection for a seaplane is the salinity of the environment. Composite materials are ideal for this application as they fulfill all stiffness and strength requirements, while being lightweight and highly resistant to corrosion and fatigue damage. Use of composite materials does however raise some additional maintainability concerns.

2.3 Seaplane Operations

2.3.1 Implications of Seaborne Operation

2.3.1.1 Effect Of Sea State on Operations

Operation of seaplanes from water presents some unique requirements in terms of their design and operational procedures. A seaplane's ability to operate from a particular stretch of water greatly depends on the local weather conditions. For open ocean operations, local wind direction should not impose any limits on operation, as takeoff direction can be adjusted accordingly, while limited heading adjustments may be possible when operation from a cordoned off area, such as a potential future seaplane airport.

As mentioned in section 2.2.3.3, the main limitation to waterborne operations is the local sea state. Large modern seaplanes have been found to perform agreeably in sea states up to 4 (2.2 m wave height). The Federal Aviation Agency (FAA) and International Civil Aviation Organization (ICAO) suggest that airports should be designed in such a way so that crosswind landings are feasible 95% of the year [22]. Similar operational requirements should be applied to seaplanes and their docking facilities. A typical probability distribution for the non-exceedance of significant wave heights is shown in figure 2.3. Data for open ocean conditions in the northern hemisphere were obtained from the US

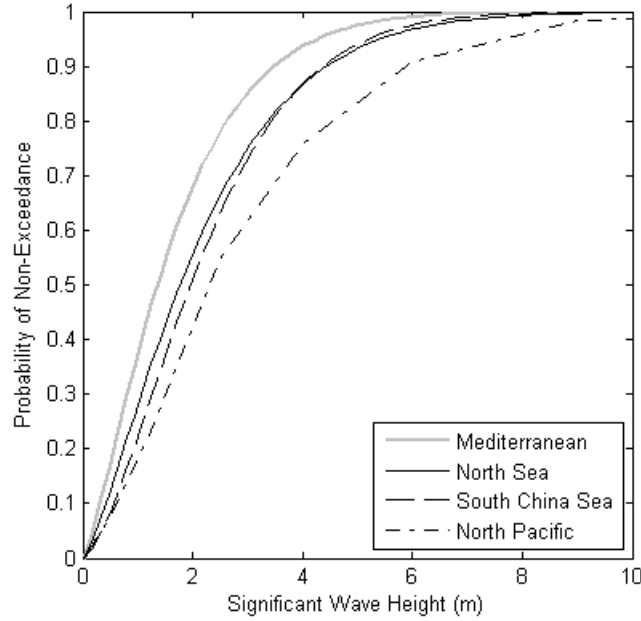


Figure 2.3: Non-Exceedance Probability of Significant Wave Height

Navy’s Spectral Ocean Wave Model [48]. Additional wave statistics were obtained from a database of shipboard observations [39], with a Weibull probability distribution having been fitted to the data to facilitate further analysis.

Figure 2.4 shows the probability that waves lower than a significant wave height of 2.5m (sea state 4) are encountered at various ocean locations around the world. Conditions in none of the regions investigated satisfied the 95% operational capability requirement set by the FAA. It is however expected that the measurements used were probably taken far offshore and conditions closer to the coast can be expected to be considerably calmer. In case wave heights encountered in shallower waters should be too high, use of wave breakers may easily remedy the problem.

This data can however be used to determine the seaworthiness requirements for SAR and cargo transport seaplanes, intended to operate as far as 2000nm offshore. In order for an operational capability of 90% to be attained in this case, a future seaplane would need to be capable of operating when significant wave heights between 3 and 5 meters (sea states 5 or 6) are encountered.

2.3.1.2 Icing

Wetting of the airframe by waves and spray, generated during takeoff, could make icing a regular occurrence, rather than one dependent on local weather conditions. Icing is the accumulation of ice

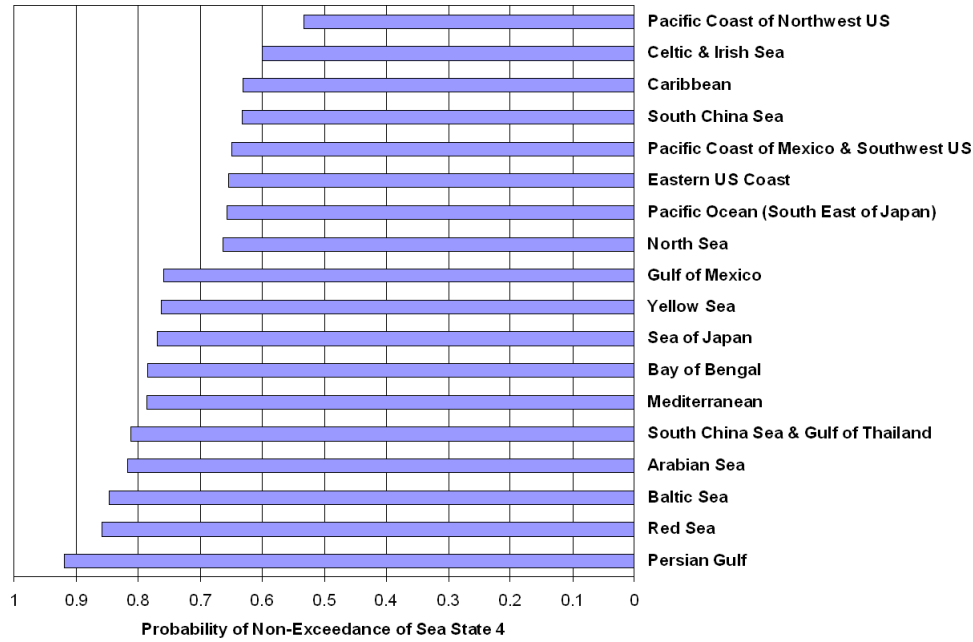


Figure 2.4: Probability of encountering waters lower than sea state 4

on the airframe. It affects an aircrafts aerodynamic performance and poses a threat to engines, as they can be struck by breakaway pieces of ice. The salinity of sea water is not likely to remedy the situation, as it freezes at -2°C ; a temperature often encountered in North America and Europe in the wintertime.

To remove ice formations, aircraft are sprayed with a heated mixture of de-icing fluids (usually glycol) and water. Anti-icing agents are subsequently applied to prevent the buildup of ice prior to takeoff. Airports are required to collect and treat these liquids as they are toxic and have been found to threaten aquatic life [90]. Therefore additional infrastructure would be required to prevent the chemicals from falling into the sea. Basins similar to a drydock could provide a contained environment in which chemicals could be used.

Some environmental impact concerns persist, as chemicals left on the airframe would still be dispersed into the water while taxiing and during takeoff. Therefore a chemical free method may be advisable. Over the past couple of years the use of Infrared Aircraft De-Icing systems has been identified as a viable alternative, offering a faster and cheaper process. Aircraft could taxi through floating hangars equipped with infrared panels, where accumulated ice and snow would be rapidly melted using infrared radiation [18].

Icing presents a challenge after the aircraft becomes airborne as well. The potential buildup of

ice on the leading edge of the wing sections can be controlled through various de-icing and anti-icing systems, widely used on current passenger aircraft. Greater concern arises from water trapped in crevices freezing, as expanding ice could help propagate cracks, compromising the structural integrity of the hull. Furthermore, icing of water trapped within the control surfaces and high lift devices could lead to a loss of control. Such concerns could be addressed by spending more time at higher speed and lower altitudes, causing trapped water to be blown out of cavities.

2.3.1.3 Attachment of Sedentary Marine Organisms

The aerodynamic performance of seaplanes can also be affected by marine wildlife. While immersed in water, sedentary marine organisms, such as barnacles or algae, attach themselves to ship hulls, increasing the surface roughness and thus fluid dynamic drag.

The fact that a potential seaplane is not expected to spend extensive amounts of time stationary on water may prevent such organisms from fouling its hull. This comes as a result of the existence of an induction period after the hull is introduced to water, when near-nil fouling rates are observed. Further, the high wall shear stresses experienced when taking off, landing, cruising on water at high speed, could clean the seaplane hull much like jets of pressurized water would. Should fouling still be an issue, other methods will have to be employed to prevent the adhesion of marine organisms or regularly clean the hull.

In naval architecture, anti-fouling coatings are used to protect the ship's submerged hull. They are usually metallic, toxic paints, however environmental concerns over their impact on marine wildlife has lead to the development of non-toxic, polymer based alternatives. The ability of anti-fouling coating to withstand the extremes of the seaplanes flight envelope should be investigated. Another alternative is having the seaplane taxi over an array of pressurized water jets that would clean the hull.

2.3.1.4 Interaction with other Vessels

The International Rules of the Road [33] regulate the interaction between vessels in inland and international waters. Under these regulations, seaplanes have to avoid impeding the navigation of all other vessels and obey the Rules of the Road if risk of collision exists. The close proximity between aircraft within the environment of a busy seaplane docking facility raises additional concerns.

The use of a ground control system similar to that employed on current airports will be required. Marking of runways and taxi channels should regulate the movement of vessels within the seaplane base, as described in section 2.3.3.2. Further, taking into consideration the impact of local water currents and prevailing winds, aircraft should not be allowed to remain stationary away from their

mooring position. Therefore a seaplane base should be designed to avoid the formation of queues near the runways, docks and mooring points.

2.3.1.5 Floating Debris

Collisions with other seagoing vessels, impact onto docks and running aground, can cause massive damage to seaplane hulls. However, as even the slightest amount of damage can compromise the seaplanes ability to stay afloat and thus complete its mission, the damaging potential of floating debris has to be investigated. Debris are likely to be a larger issue in the marine environment, as wood and other buoyant materials could simply drift into a seaplane base.

After Foreign Object Damage was found to be the underlying cause of the Concorde crash in 2000, there has been increased research into runway debris detection systems, such as QinetiQ's Tarsier system. Similar systems could be used to locate debris on rough waters and dispatch a cleaning crew, allowing operations to continue with minimal disruption. It should be noted, that unlike aircraft operating from paved runways, seaplane takeoff headings could be adjusted to avoid the debris until it is retrieved. Seaplanes with mission profiles requiring that they operate from the open ocean, should be equipped with appropriate radar equipment, enabling the detection of drifting debris, sizable enough to damage the hull.

The worst case scenario comes in the form of a plane crashing and disintegrating in the middle of the "runway". In such a case large amounts of floating debris will be left floating. Seaplanes could still be allowed to operate from areas further seaward and taxi to the coast, thus not totally disrupting services. In order for the takeoff and landing area to be cleared, boats equipped with nets could be used to contain and clear the debris swiftly.

2.3.1.6 Bird Hazards

Coastal areas often provide refuge to a variety of birds, which rely on marine wildlife for sustenance. The increased presence of seabirds may increase the likelihood of a bird strike and thus hinder seaplane operations in certain regions. Various methods have been developed to reduce the risk of bird strikes in areas surrounding airports, yet no single recipe for success has been found as a particular method's effectiveness largely depends on the peculiarities of each situation.

The most suitable methods for use at a seaplane base could be identified through case studies of the bird control programs currently implemented by certain large airports in coastal environments [36, 96].

2.3.1.7 Safety Equipment

As stated in section 2.2.5, airworthiness requirements mandate that seaplanes have a number of watertight compartments, to keep the aircraft afloat in case of a hull breach. Yet, in case of an emergency passengers and crew should be able to swiftly evacuate an aircraft. Therefore any potential seaplane should be equipped with emergency exits and flotation devices. Ditching requirements for current landplanes could be taken as a guide. As specified by FAR 25.807, provisions will have to be made to place emergency exits above the waterline and prevent flooding of the passenger cabin. The exit height should be further adjusted for the possibility of numerous watertight compartments flooding and rough seas. Floating escape chutes, similar to those used on current aircraft in case of ditching, can be used as life rafts.

2.3.2 Maintenance

Seaplanes, like landplanes, will have to meet airworthiness and operating standards set by regulating bodies such as the ICAO, the Joint Aviation Authorities (JAA), the FAA and the European Aviation Safety Agency (EASA). One of the requirements is that aircraft undergo maintenance checks at regular intervals, depending on its age, flight time and the number of takeoffs and landings it has performed. Even though the exact maintenance schedules and procedures used by manufacturers and airlines differ, the maintenance program outlined in table 2.2 is indicative of the procedures followed by most operators.

2.3.2.1 Inspection of Submerged Parts

The lack of access to submerged parts of the aircraft, is likely to hinder the work of inspectors while the seaplane is docked. Bringing the seaplane onto shore for all inspections, would be very time consuming and severely impact operating costs. Further, requiring that divers inspect the hull would be unreasonable, especially when adverse weather conditions are encountered. As only visual inspection of the hull is required, waterproof cameras could be moved under the aircraft to assist the inspector.

The visual inspection requirements could also be somewhat relaxed by installing autonomous damage detection systems to monitor the structural integrity of the hull. Inspection of the hull could still be done using robots, equipped to perform underwater non-destructive testing. Similar systems are currently in use for inspecting underwater pipelines.

Name	Interval	Description
PS-Check	2-3 days	Visual inspection at gate. <i>2 man hours</i>
A-Check	80-100 flight hours	More detailed than PS-check usually held at gate overnight. <i>10-20 man hours</i>
B-Check	500-600 flight hours	Aircraft moved to hangar to be serviced and for specific systems to be checked. <i>100-300 man hours</i>
Widebody C-Check	24-30 months	Complete inspection and overhaul of the aircraft. <i>10,000 man hours</i>
Narrowbody C-Check	15-18 months	Exhaustive set of inspections and overhaul of systems and the entire airframe done in hangar. <i>2,100 man hours</i> Every fourth check is more detailed taking <i>20,000-30,000 man hours</i>

Table 2.2: Maintenance Schedule of The American Airlines Fleet [7]

2.3.2.2 Beaching

The aircraft will eventually have to be moved out of the water and into a hangar for deep maintenance. Several ways of removing aircraft from the water, such as dry docks, cranes and winch systems, were investigated. The use of a dry dock or cranes was deemed less likely, as large capital investment would be required and such a system would be difficult to use with very large aircraft. Use of a winching systems was also found inappropriate, as lack of wheels means that the hull may get damaged and maneuvering the aircraft on land would be difficult.

The most effective beaching system found is a cradle equipped with wheels. One such apparatus, seen in figure 2.5, was developed and patented by the Martin Company for the P6M SeaMaster [37]. A seaplane would taxi over the semi-submerged cradle, which would in turn secure itself to the airframe. The seaplane could then taxi up a marine ramp under its own power. The ramp's incline would depend on the aircraft's thrust to weight ratio and vice versa. Giving the pilot control over the rotation of



Figure 2.5: P6M SeaMaster Taxiing out of the Water

the cradle's wheels would allow the seaplane to easily taxi on land, while freely rotating wheels could prove quite useful for maneuvering the aircraft inside a hangar. Even though each seaplane type could use bespoke beaching system, adjustable cradles could be designed for use with multiple aircraft types and sizes.

A breasting-in rig is also part of the apparatus patented by the Martin Company. The rig is comprised of two long swiveling spars, set so as to form a V-shape. A retarding cable is attached at the tips of the spars, spanning between them. When this cable is grappled by a retractable hook on the underbelly of a passing seaplane, the aircraft is slowed down and the tips of the two spars are forced together. In turn, the spars form a channel correcting the seaplanes heading. Such a rig could prove very useful not only in conjunction with a beaching cradle, but to keep seaplanes from drifting onto stationary objects. One very likely application is in guiding seaplanes into their stands (gates) while avoiding contact with neighboring parked aircraft.

2.3.3 Seaplane Base Design

2.3.3.1 Requirements for Water Area

The design of seaplane bases is regulated by the FAA under Aviation Circular 150/5395-1 [59]. Although dimensions given in this document should be treated as the bare minimum requirement, since it refers to light aircraft, it should still be used as a guide for the layout of a commercial seaplane base.

Seaplane pilots prefer to use unmarked sea lanes for takeoffs and landings, so as to adjust their orientation for waves, winds and currents. Unfortunately, such freedom cannot be allowed within the crowded environment of a commercial seaplane base. Sea lanes (takeoff and landing areas) should

be situated where local currents are less than 5.5 km/h and aligned so as to provide maximum wind coverage. The height and location of potential obstacles to air navigation should be regulated per Annex 14 of the ICAO. If multiple runways are to be used in parallel, their separation should be such that air turbulence and water waves generated by one aircraft do not hinder the takeoff or landing of another. The use of a wave breaker or situating the terminal between the two parallel runways may serve to minimize interaction between two aircraft simultaneously taking off or landing.

It is recommended that taxi channels (taxiways) provide direct access to the onshore facility (terminal building) and if possible face into the prevailing wind or current. Appropriately sized clearance should be left between the edge of a "runway" or taxi channel and the nearest obstruction.

2.3.3.2 Marking and Lighting

In order for navigation within a seaplane base to be possible, operating areas (runways) and taxi channels will have to be clearly marked. Further, in order for operations after dark to be possible, the operating areas and taxi channels must be appropriately illuminated.

The IALA Maritime Buoyage System can be used to direct seaplanes to and from the operating areas, mark junctions of taxi channels and separate outgoing and incoming traffic. Buoys can also be easily illuminated using colored and pulsating lights. Although the same buoy shapes are consistently used worldwide, opposite colors are used for buoyage systems in the Americas, Japan, Rep. of Korea and the Philippines (region B) and the rest of the world (region A). In region B, shown in figure 2.6, green and red buoys are used to mark the left and right sides of channels accordingly for vessels are traveling from seaward. Junction buoys instruct a vessel of the preferred channel, when a junction is encountered, i.e a port junction buoy must be kept to the left of a vessel when coming from seaward. Fairway buoys should be kept to the port side of all vessels and are thus used to separate upstream and downstream traffic.

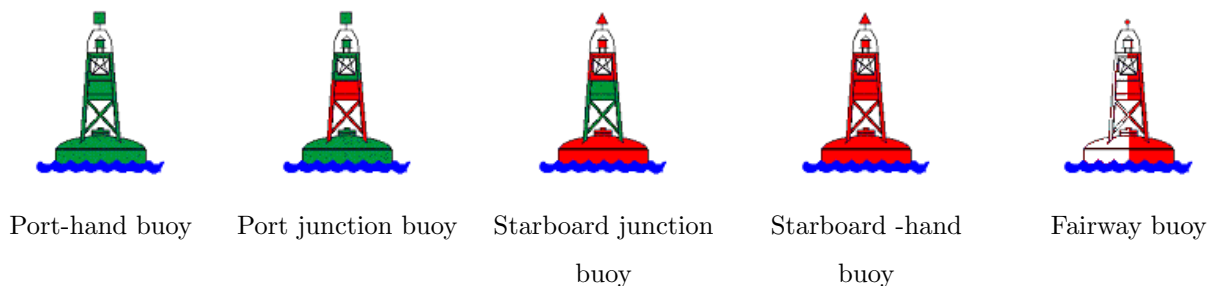


Figure 2.6: Buoyage used to guide vessels in region B

Lighting and marking the operating area poses a larger challenge. The aeronautical lighting and visual aides used by airports are described in the FAA’s Aeronautical Information Manual [30]. As the seaplane base is likely to be used by large commercial aircraft, the operating area should be treated as a precision instrument runway. A number of lighting systems are required to inform the pilot of the specific points on a runway, its orientation and its boundaries at night and under adverse weather conditions. The Approach Landing System (ALS), used to assist pilots in transitioning from instrument flight to visual flight, starts at the landing threshold and extend into the approach area. Runway End Identifier Light (REIL) and Runway Edge Light Systems are also required to inform pilots of the boundaries of the runway. These three lighting systems can be can be placed above the water surface as a seaplane is not likely to approach them. In fact appropriate clearance should be left so that their appearance is not altered by waves.

This is not the case for in-runway lighting, such as the Runway Centerline Lighting System (RCLS) and Touchdown Zone Lights (TDZL). These systems will have to be submerged so that seaplanes can take off and land over them without damaging their hull. Submerged lighting should also be considered for lead-on and lead-off lights used to guide aircraft from taxi channels into the operating area and vice versa. However, submerging the lights will result in certain optical effects by the water, raising concerns over the image perceived by pilots being distorted.

Several markings are also required along the runway to inform pilots of the runway threshold, designation and centerline. Markings for the runway ”touchdown zone” and ”aiming point” are also required. As these cannot be painted onto water, submerged lighting could be used instead; although its effectiveness in daylight conditions should be investigated. As far as the designation marking is concerned, there is also the possibility of painting it on a floating platform ahead of the operating area’s threshold.

2.3.3.3 Airport Boundaries

Section 2.3.1.1 identified local wave conditions as a major limiting factor for seaplane operations. Furthermore, as noted in sections 2.3.1.5 and 2.3.1.4, the impact of a seaplanes hull with floating debris or another vessel can be catastrophic. Thus means of isolating the sea base area from the rest of the sea are required.

Using an offshore breakwater, wave heights can be decreased sufficiently for uninterrupted operation of seaplanes. The location and orientation of the breakwater will depend on the local wave characteristics. Access into the seaplane base to all non-authorized vessels should be restricted, in

order for the likelihood of obstructing or colliding with seaplanes to be minimized. This could easily be achieved by fencing off the area. Similar damage could be done to a seaplane hull as a result of colliding with a whale or the shell of a sea turtle. Therefore, the use of submerged netting or repellents to keep such species away may have to be considered.

2.3.3.4 Mooring and Docking

Contact Stands

Civil airliners are usually boarded at a contact stand, commonly referred to as a terminal gate. When boarding, refueling and loading or unloading numerous structures are placed onto or in very close proximity to the aircraft. Therefore, as floating objects tend to move under the influence of waves and wind, seaplanes will have to be fastened in place to prevent their hull from getting damaged. The fluctuation of the water level, as a result of the tides, must also be taken into consideration when choosing a mooring system.

The simplest way of securing a seaplane is through use of multiple mooring lines. The lines should subsequently be taut or given slack according to the fluctuations in water level. An alternate automated system, that could be modified to allow safer and faster mooring for seaplanes, is vacuum mooring [58]. The suction generated by vacuum pads attached onto the seaplanes airframe would secure the aircraft in place. However, given the high suction forces required and the fragility of aircraft airframes, the applicability of such a system to mooring seaplanes will have to be investigated.

Once the seaplane is secured in place, multiple jet bridges and ramps can be extended to the seaplane in order for passengers and cargo to be loaded or unloaded. A submerged distribution system could be used to supply fuel and potable water to the aircraft and service the lavatories. Offshore Petroleum Discharge Systems (OPDS), currently used for the transfer of petroleum from a tanker to the shore, are an example of such a distribution system.

Remote Stands

As civilian seaplanes are not expected to carry anchors due to the associated weight penalty, anchorage areas will have to be designated so that seaplanes can be parked away from the terminal building. Rigidly mooring a seaplane to a buoy can severely damage the airframe, as high snap loads can be developed in high wind conditions. Instead, the use of an "anti-snatch" mooring system has been suggested for use by the US Navy's Sea Base [62]. Thus a restoring force would be generated as a function of displacement, not subjecting the seaplane to an otherwise very violent loading.

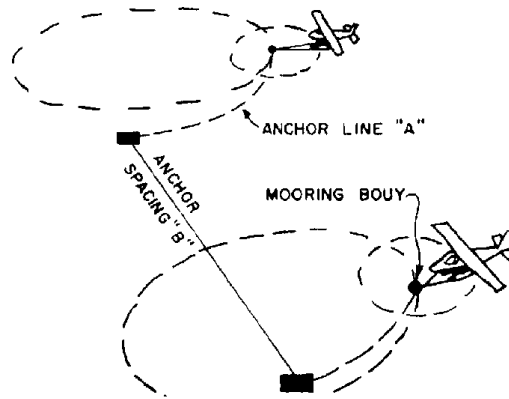


Figure 2.7: Light aircraft moored on two adjacent buoys [59]

The number of moorings that can be installed will depend on the area available, the maximum aircraft size to be accommodated and the depth of the water. The FAA [59] requires that the length of the anchor line, "A" (figure 2.7), be no less than six times the maximum depth of the water. This length requirement may be halved if the anchor's holding capacity is doubled. Further, the spacing between two adjacent anchors must be at least twice the length of the longest anchor line plus a minimum of 70m for larger aircraft.

2.4 Environmental Impact

2.4.1 Wildlife

Environmental concerns have become a major factor in the design and operation of civil aircraft. Aircraft noise and the harassment of bird populations around airports, can be identified as the main impacts of aviation on wildlife. The waterborne nature of seaplanes would give rise to concerns over the impact of aviation on marine wildlife as well.

One major environmental concern in the shipping industry is the transport of non-native/invasive marine organisms. Various marine species can be transported from one port to another inside a ship's ballast water tanks or by attaching themselves to a ship's hull. Should these non-native species have no natural predators in the new environment, their numbers increase displacing or killing other native species. The local ecosystem may also be harmed by the introduction of foreign pests, parasites and diseases. In the past fouling on flying boat hulls and anchors has been blamed for the introduction of foreign organisms to the British marine ecosystem [26]. Unlike older seaplanes, modern ones will cruise at much higher altitudes and speeds, reducing the likelihood of organisms surviving on their hull and removing any water trapped in crevices. Should organisms be found to be resilient to the

adverse conditions experienced during cruise, the use of toxic anti-fouling coatings may have to be considered.

The possibility of fuel and oils contaminating the water in the event of a crash or leak poses another concern. Kerosene (jet fuel) in particular is highly toxic to both man and wildlife. Therefore, methods of containing a possible fuel or oilspil within the bounds of a seaplane base will have to be investigated.

2.4.2 Noise

Aircraft generated noise is the most noticeable environmental effect of aircraft, particularly for populated areas surrounding airports. As no significant reductions in noise are expected, the solution is to move the problem away from most people. Figure 2.8 shows that noise is most severe along an aircraft's takeoff and landing path. Use of seaplanes would shift these paths over the water, keeping populated areas out of an seaplane base's noise contours.

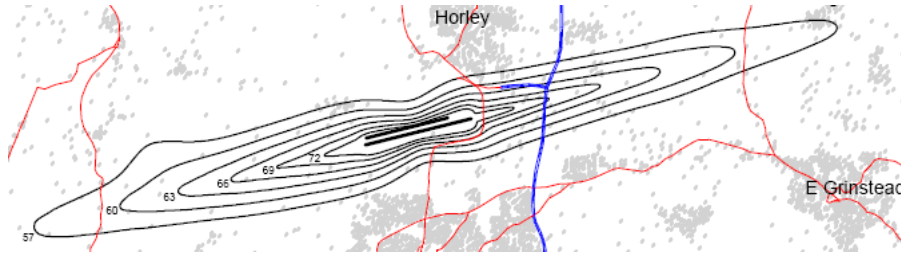


Figure 2.8: Noise contours around Gatwick Airport. [57]

A large proportion of aircraft noise can be attributed to turbofan engines. As the operational characteristics of a BWB seaplane require that engines are placed on top of the fuselage, the airframe would effectively stop some of the engine noise from reaching the ground.

Recently there has also been increased concern over the effect of ocean noise on marine wildlife and marine mammals in particular. Although the amplitude of the noise is expected to be less than that for conventional marine vessels using submerged propellers for propulsion, an investigation into the noise generated by seaplanes when planing at high speed may be prudent.

2.4.3 Fuel Consumption and Emissions

The considerably higher lift to drag ratios of flying wing aircraft implies that the same weight can be lifted using a much lower amount of thrust. Therefore, advanced engines and lower thrust requirements lead to reductions in both fuel consumption and emissions.

However, the marine environment that seaplanes will operate in does raise some issues. Under the right conditions, ingestion of salt (NaCl) into an engine may produce $\text{C}_{12}\text{H}_4\text{O}_2\text{Cl}_4$ as a combustion product [3]. Additional research into the combustion of engines in the presence of salt is highly recommended, as this dioxin is a known carcinogen to humans and may thus endanger passengers, staff and nearby residents.

Chapter 3

Literature Review

3.1 Introduction

The literature relevant to the design of flying boats is addressed in this section. In addition to key texts addressing the holistic design of both landplanes and flying boats, certain aspects of aerodynamic design relevant to flying boats are addressed. These include the aerodynamic drag prediction for stepped hulls and the performance of wings in very close proximity to the ground. A brief review of the key advances in hull hydrodynamic design is presented, along with some modern computational and empirical methodologies that could be implemented when designing the seaplane hull.

3.2 Aircraft Design

Several core texts address the initial and preliminary design of aircraft. Numerous empirical and theoretical methodologies, best practice guidelines and a wealth of data on existing aircraft are presented by Raymer [67] and Roskam [69, Vol. 1-8]. Within this text, the USAF's DATCOM method and several methods developed by Torenbeek [93] and L.M. Nicolai are also presented. Methods for preliminary aerodynamic, structural, stability and performance analyses are presented, however their applicability is somewhat limited to conventional configurations of light, transport and fighter aircraft. Further, the design of waterborne aircraft is briefly discussed qualitatively, with little quantitative information being provided.

The design of seaplanes is examined in more detail by Nelson [61] and Langley [47]. Although antiquated, these texts are a good starting point, presenting the requirements for waterborne static stability and the basic hydrodynamic theory required to predict impact loads.

The advances in flying boat design over the 1940's were later presented by Stout [89]. Emphasis was placed on hull design and hydrodynamic testing. The effects of increasing a hull's length to beam ratio, such as the resulting reduction of hull resistance and spray severity, are discussed. The steps required to broaden the trim limits for a stable takeoff are also addressed.

More recently, Stinton [88] presented a concise summary of past seaplane design experience. Although mostly qualitative, several best practice rules for seaplane design are presented. Moreover, the operational peculiarities of operating from water are described. Another review of past research into past seaplane research, by Hamilton and Allen [35], examines the contribution of the Marine Aircraft Experimental Establishment to seaplane design, presenting data from several MAEE reports on seaplane waterborne stability, step design and airborne drag.

In the past decade there has been an increased interest in next-generation seaplane design by the US Navy, leading up to its Sea Basing concept [24]. The potential of using seaplanes for rapid strategic airlift and their integration with a Sea Base was investigated by Odedra et al. [62]. A parametric study of past seaplane designs is presented, leading to the conceptual design of a 160,000 lbs seaplane of conventional design. Similarly, Bellanca and Matthews [8] designed three seaplanes, weighing 0.3, 1.0 and 2.9 million pounds, to quantify the effects of maturing technology on seaplane performance. Use of composite materials and employing a retractable step fairing both resulted in considerable performance improvements. However, in both reports only seaplanes of conventional design were considered.

In 1993, an investigation into Russian wingship design knowhow was commissioned and reported by the Advanced Research Projects Agency [3, 4, 5]. A thorough review of the state of the art in extreme ground effect aerodynamics and hydrodynamic design was done. Although more than a decade later, the finding in these two areas can be considered current, given the limited advances reported in those two fields. Of particular interest is the section referring to rough water operations, giving formulae for impact load estimation.

Information on the design and performance of various aircraft is also available. The development of Canadair's CL-215 amphibian, is presented by Remington [68]. The aerodynamic and hydrodynamic design is discussed and performance data is presented. Further, the seaplane design process currently employed can be better understood, as the reasons behind several design choices are often mentioned or explained.

The advantages of the Blended-Wing-Body or Flying-Wing over conventional configurations have lead to a number of research programs over the past two decades. Liebeck [49] presented the results of a joint program between NASA and Boeing. In the European Union a great deal of research has been undertaken as part of the MOB project. Parts of this work have been presented by Mialon et al. [55] and Qin et al. [66]. The research efforts by TsAGI in Russia were also made public by Bolsunovsky et al. [12], addressing a number of issues associated with the design of flying wings.

3.3 Aerodynamic Design

The aerodynamic methods used for designing and analyzing flying boats are mostly identical to those widely used for similar land based aircraft. The main difference arises from the shaping of the seaplane's underbelly, which features hard chines and backward-facing steps to meet hydrodynamic performance requirements.

The aerodynamic drag of flying boat hulls is addressed by Hoerner [38]. The incremental drag penalty of using hard chines is shown to be comparatively less than that associated with the use of steps. For the estimation of the drag caused by rearward-facing steps, Hoerner suggests that a method intended for two-dimensional steps immersed in a turbulent boundary layer is used. This method however only applies for step heights smaller than 0.9 of the local boundary layer thickness and was found to under predict the drag force by up to 50%, when compared to experimental data. Another such method is given in ESDU 75051 [27] but is only applicable for step heights less than 0.1 times the local boundary layer thickness.

Steps and chines are often faired to improve the aerodynamic performance of flying-boat hulls. Figure 3.1 illustrates several different fairing designs, of varying complexity, and their effect on hull surface drag as presented by Smith and Allen [85]. Minimum drag was observed using a straight 9:1 fairing, reducing step related drag by approximately 83%.

Investigating the effects of backward-facing steps on the aerodynamic characteristics of a NACA 0012 airfoil, Finaish and Witherspoon [31] observed changes not only in drag, but also on the airfoil's lifting characteristics. The case where a step is placed on the airfoil's bottom surface, extending from mid-chord to the trailing edge, bears particular interest. At an angle of attack of zero degrees, an increase in lift-to-drag ratio proportional to step height was observed. Increasing angle of attack was found to reduce the effect of step height, with in lift-to-drag ratio becoming inversely proportional to step height for angles of attack above five degrees.

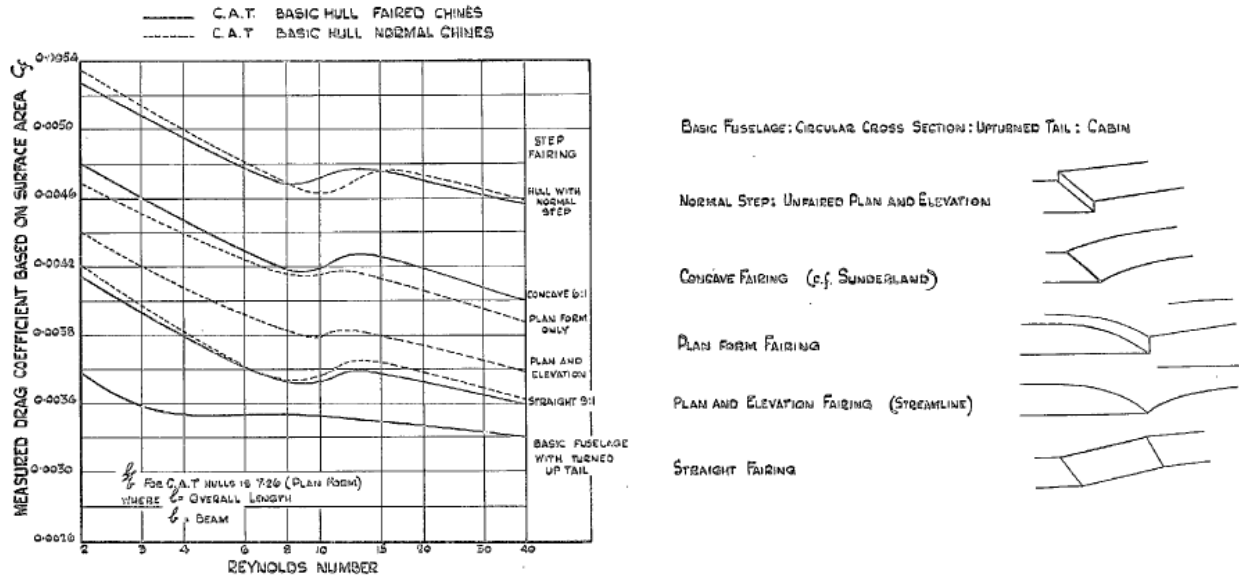


Figure 3.1: Effect of different fairings on hull drag [85]

During takeoff and landing, the aerodynamic effects of extreme proximity to the ground will also have to be considered. The implications of flying at a height which is lower than the wing's chord are thoroughly presented by Rozhdestvensky [75]. Lifting surfaces are found to experience an increase in lift-to-drag ratio, while the airfoil's center of pressure moves between $1/3$ and $1/2$ chord increasing the nose down pitching moment [94]. These effects can severely impact the aircraft's stability during takeoff and landing. Additionally for the analysis of high aspect ratio wings in extreme ground effect, Rozhdestvensky [76] has presented a modified lifting line method.

3.4 Hull Design

When designing the hull of high speed craft, all gravitational, aerodynamic and hydrodynamic forces acting on the vessel have to be considered simultaneously, however their magnitudes greatly vary with the velocity, attitude and draft of the craft. This complexity inherent to operating on water-air boundary has led to a relative lack of simple, accurate methods for hull design. Instead naval architects have historically relied heavily on tank tests to determine the seakeeping characteristics of hull forms.

High speed marine vessels operate in three regimes. At low speed, the displacement regime, the majority of lift experienced by the craft is a result of buoyancy, a hydrostatic force proportional to the vessel's immersed volume. The semi-displacement regime is encountered when speeds become high enough that a mix of hydrodynamic and hydrostatic forces are applied on the vessel. As speed

increases further, the vessel enters the planing regime. When planing the hydrodynamic pressure loads become large enough to support the vessel's weight, reducing its immersed volume and thus the hydrostatic forces.

The steady planing behaviour of a prismatic hull can be predicted by analysing the water impact of its cross section at various stations along its length, as described by Faltinsen [29]. The earliest theory on 2-D impact of wedges in water was formulated by von Karman [97], assuming the effects of gravity are negligible and ideal flow. Wagner [98] later modified this theory to account for the effects of water rising on the sides of a wedge. Observing that Wagner's model persistently overestimated wedge impact loads, Mayo [53] re-derived the general equations to take longitudinal flow effects into consideration. This model was extended for use on non-prismatic seaplane hulls by Milwitzky [56], primarily considering the impact of scalloped-bottom seaplane hulls.

More recently use of modern computing resources has led to some novel approaches to the issue. The water entry of wedges was investigated by Zhao and Faltinsen [100] using both a boundary value problem formulation and a similarity solution. Results obtained using both methods compared very well to experimental data. Another boundary value problem formulation was presented by Savander et al. [77], allowing the analysis of non-prismatic three-dimensional hulls. Predictions of the resistance, trim angle and draft of Series-62 hulls compared extremely well with experimental data at high speed, however no comparison was made for volumetric Froude numbers (F_{∇}) lower than 2.5.

In addition to the work available for the theoretical prediction of planing hull impact loadings, several semi-empirical design methods have been developed. A methodology for the complete hydrodynamic design of planing hulls is presented by Savitsky [79]. Equations for predicting hydrodynamic forces, moments and hull wetted area were developed using towing-tank test data for prismatic planing surfaces. This methodology was later supplemented with relations for the prediction of hydrodynamic resistance in the pre-planing range by Savitsky and Brown [78] and the drag generated by whisker spray by Savitsky et al. [80].

An alternative empirical formula for estimating the hydrodynamic lift of planing surfaces has been suggested by Shuford [84]. Very good agreement with experimental data was observed, particularly at higher wetted length-to-beam ratios for which the formulae quoted by Savitsky are not applicable.

In addition to the texts cited above, there is a wealth of experimental data, stemming from almost 50 years of seaplane research by the National Advisory Committee for Aeronautics (NACA) and the

Aeronautical Research Council (ARC). Information from multiple such reports was summarised by Dathe and de Leo [20], illustrating the effect of various hull shape parameters on the hydrodynamic performance of seaplanes.

Dynamic instabilities experienced by planing hulls at high speed are a major concern, as they can result in hull damage and inconvenience passengers. Chief amongst them is porpoising, a combined pitch-heave instability that forces flying boats to operate in a narrow band of trim angles during takeoff. In a review of experimental results available up to the early 1950's, Smith and White [86] provides some guidelines for the aerodynamic and hydrodynamic design of stable hulls. More recently Celano [17] presented an empirical equation for the prediction of the critical trim angle associated with the onset of porpoising on prismatic hulls. In the case of non-prismatic hulls it may be possible to predict the inception of porpoising using the method presented by Martin [51].

Chapter 4

Baseline Configuration & Initial Sizing

4.1 Baseline Configuration

The baseline configuration details the general arrangement of the proposed design. This initial sketch is the result of qualitative considerations for meeting a set of predefined specifications and identifies the general shape of the aircraft and the location of major components or system groups. It is used as a guide in the development of the aircraft synthesis algorithms, however the final optimised aircraft shape and configuration is often different in order for mission requirements to be met.

4.1.1 Baseline Design Justification

The aim of the baseline design detailed in this chapter is to mitigate some of the problems associated with designing aircraft to operate from water. Experience with previous seaplane designs has shown that, in comparison with land planes of equivalent size, they suffer from both increased structural weight and aerodynamic drag.

Both a monohull and multihull (catamaran) configuration were initially considered. Although the catamaran exhibits better lateral stability characteristics on water, the monohull results in a lower increase in wetted area therefore resulting in a lower drag penalty. There is also evidence that using a catamaran configuration could adversely affect the aircraft's directional stability. Finally, a monohull can be better integrated and blended with the rest of the aircraft, reducing the amount of interference drag that would arise. The exact hull design and its dimensions cannot be determined at this early stage, as they will heavily depend on each individual aircraft's aerodynamic, propulsive, weight and balance properties. As discussed in section 3.3, the use of a retractable fairing on the main step could further reduce aerodynamic drag during takeoff and landing.

Given the choice of a monohull design, lateral hydrostatic stability becomes a major concern. In the past, lateral static stability has been enhanced using tip floats, resulting in substantial weight and aerodynamic penalties for seaplanes. The use of sponsors or seawings that could enhance lateral stability and ensure excessive roll angles are avoided was briefly considered, however the resulting increase in wetted area was unpalatable. Instead the idea of using part of the wing's root to act as a seawing was found to both not increase the aircraft's wetted area and ensure that sufficient righting moments are produced at excessive roll angles. Moreover it was felt that the larger the chord of the seawing used, the less strengthening would be required to ensure hydrodynamic impact loads are withstood, thus further reducing the weight penalty. This idea was successfully demonstrated by Serghides [81] for the Advanced Amphibian Water Bomber concept aircraft, designed in 2008.

This hydrostatic lateral instability is also a result of using high length-to-beam ratio fuselages and the height centre of gravity arising from the need to locate several heavy components high above the waterline. Therefore by blending the hull onto the wing, as seen in the Consolidated Vultee Skate aircraft and the Beriev Be-103, or using a lower length-to-beam ratio hull the metacentric height can be increased, improving roll stability by lowering the centre of gravity and increasing the width of the waterplane area. Such a move could also reduce the required seawing span.

To avoid excessive loads, the wing tips must be kept clear of the water at all times. This is achieved by mounting the outer wing high on centre section and utilising sufficient levels of dihedral. If the thickness of the centre body is insufficient for a high mounting, use of a gull wing configuration is necessary.

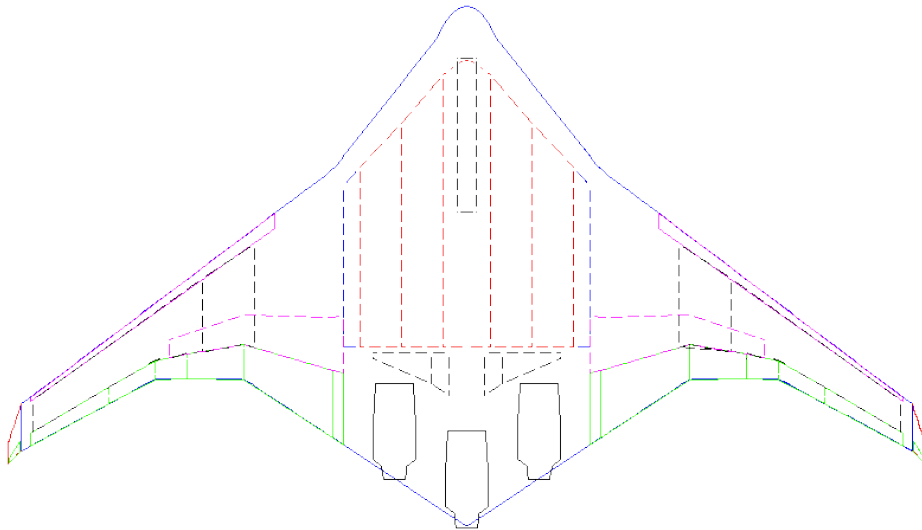


Figure 4.1: Top-view of the baseline aircraft

The aforementioned marine design considerations were found to lead to what is effectively a Blended Wing Body configuration, whereby payload is carried in a streamlined central wing section which is seamlessly blended with the outer wing. Blending the hull bottom into the centre section, the resulting body should remain streamlined with minor increases in thickness and wetted area. The resulting aircraft should therefore exhibit substantially lower levels of aerodynamic drag and reduced structural weight.

The engines will be placed on top of the fuselage, near its trailing edge. Their position will allow them to operate clear of any spray generated during takeoff and landing. To minimise the nose down moments that a high mounted engine would generate in flight, the use of boundary layer ingesting propulsion systems would be ideal. However due to the relative immaturity of such technologies, high bypass ratio turbofan engines were chosen instead, considering the aircraft's intended mission and their lower fuel consumption. In future designs turbofans could be easily substituted by prop-fans, leading to even greater fuel savings.

4.1.2 General Arrangement

The general arrangement of the aircraft resulting from the considerations presented in section 4.1.1 can be seen in figures 4.1 and 4.2. The colour and line style used to represent each different component are:

Airframe structure	Solid blue
Engines	Solid black
Fins	Solid red
Cabin bays	Dashed red
Cargo bays	Dashed blue
Fuel tanks	Dashed black
High-lift Devices	Solid Magenta
Control Surfaces	Solid green
Spoilers	Dashed Magenta

As in previous BWB designs, passengers are seated in a number of adjoint single or twin aisle cabin sections. To abide by safety regulations, the cabin can only be placed above the waterline at maximum load. Cargo bays can be situated either outboard of the cabin section or underneath it. If cargo is stored underneath the cabin, and therefore beneath the waterline, provisions must be made

to allow cargo containers to be lowered through the cabin during loading/unloading.

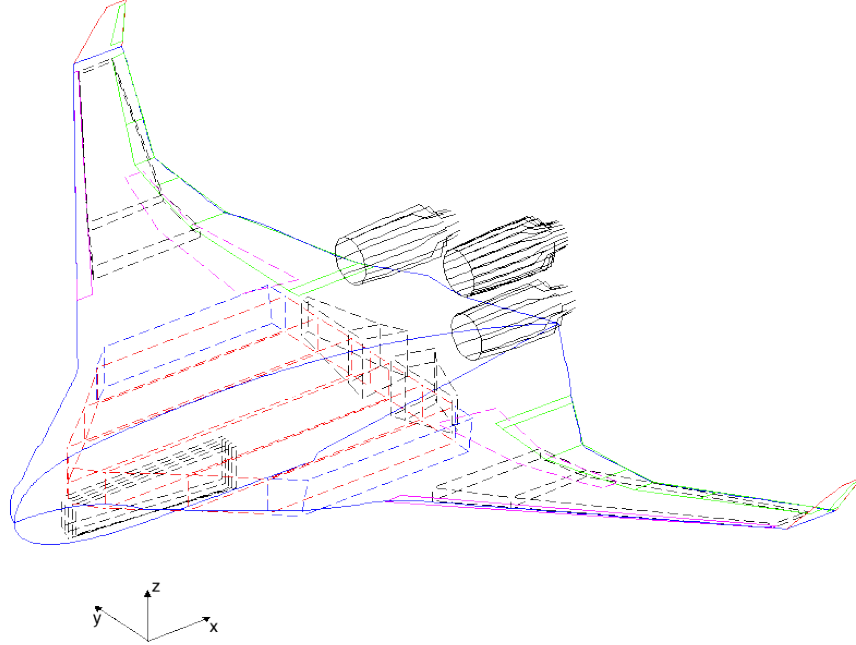


Figure 4.2: Isometric projection of the baseline aircraft

The absence of large vertical and horizontal stabilisers is advantageous, as it leads to reductions of structural weight as well as the frictional and interference drag components. Lateral stability is maintained by mounting the wings in a high position in addition to the use of dihedral, sweepback and the placement of small fins on the wing tips. The use of tip fins should also improve the aircraft's performance by reducing the lift induced drag generated. Lateral control is achieved by placing drag rudders, a splitting control surface designed to generate drag, on the fins and the outboard wing sections if necessary. Elevators and elevons are placed along the rest of the wing's trailing edge, outboard of the engines, allowing control in roll and pitch.

In order for the aircraft to be statically stable longitudinally, a planform whose centre of pressure is aft of the aircraft's centre of gravity under all loading conditions must be identified. This arrangement however, coupled with the nose down moments generated by the propulsive units, requires negative elevator deflections to maintain trim. Elevator deflections, and the resulting trim drag penalty, can be minimised if the centre of gravity coincides or remains near the wing's aerodynamic centre.

The centre of gravity position can be controlled by pumping fuel around the three fuel tanks defined in figure 4.1. The main fuel tanks are placed integrally within the outboard wing. Additional

fuel tanks are situated underneath the cabin within the hull's fore-body and behind the cabin. The placement of fuel within the hull, underneath the waterline is particularly advantageous as it can lower the centre of gravity, improving waterborne lateral stability

The lack of horizontal stabilisers also hinders the use of flaps during takeoff and landing, as the nose down moments generated when actuated cannot be effectively countered. Instead, slats or alternate leading edge high lift devices are used to increase lift and the stall angle of attack during takeoff and landing.

If lateral instabilities are identified or enhanced manoeuvring capabilities are desired, a pair of stabilisers placed near the trailing edge, outboard of the aircraft's engines can be used. Controllability could improve if ruddervators are employed. Their use will lead to a definite increase in both weight and friction drag, however trim drag during cruise may be minimised while maintaining static stability.

4.2 Initial Sizing

In the initial stages of the design process, one must determine the aircraft's maximum takeoff weight, its wing area and maximum thrust required. This initial estimate is based on the fuel required to meet a specific mission profile, representative of the aircraft's main mission requirements. The empty weight of an aircraft capable of carrying the required payload and fuel is determined using empirical relations, based on past aircraft of similar design.

4.2.1 Mission Profile

Defining an aircraft's mission profile is instrumental in quantifying the amount of fuel an aircraft must carry to fulfil its mission. A passenger or cargo carrier must be capable of cruising a certain distance, at transonic speeds and high altitudes. At the end of the cruise segment a given time duration is reserved for loitering ahead of landing. For land based aircraft it is standard practice for an additional diversion cruise segment to be included to account for unexpected events. Although a waterborne aircraft may not have to cruise far to find an alternate landing site in case of emergency at the airport, this diversion segment should be included in case of adverse weather conditions. A typical mission profile for a passenger aircraft is shown in figure 4.3 and each mission segment is described in table 4.1.

The various mission segments defined can be rearranged in order for an aircraft to be designed

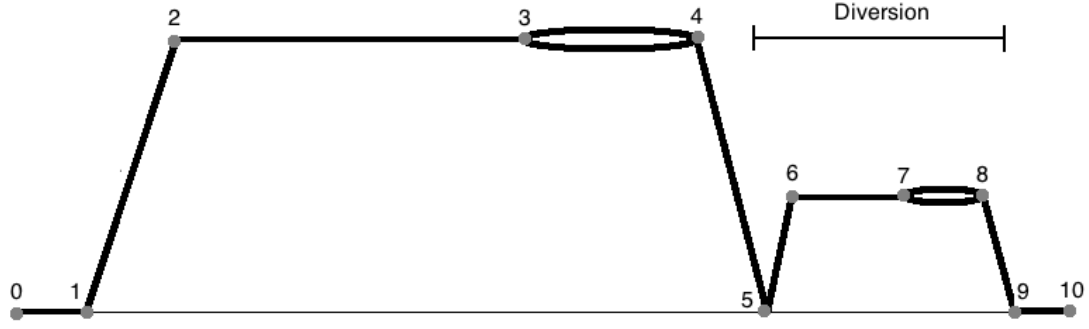


Figure 4.3: Mission profile for long range airliner

Segment	Description
0 - 1	Taxi to runway and Takeoff
1 - 2	Climb & Accelerate to cruise conditions
2 - 3	Cruise for given range
3 - 4	Loiter at cruise altitude
4 - 5	Descent fro landing and perform missed approach
5 - 6	Climb to diversion cruise altitude
6 - 7	Cruise to alternate landing site
7 - 8	Loiter
8 - 9	Descent to alternate landing site
9 - 10	Landing & taxi to gates

Table 4.1: Description of mission segments shown in figure 4.3

for more complicated mission profiles, such as maritime patrol or Search & Rescue. Additionally, if a payload drop segment is defined, aerial tankers, water bombers or minelayers can be designed.

4.2.2 Initial Weight Estimation

Having defined the aircraft's design mission, the aircraft's Maximum Takeoff Weight (MTOW) can be estimated based on the required payload, fuel and aircraft empty weight.

$$W_o = W_e + W_{pld} + W_f \quad (4.1)$$

Payload weight is specified by design requirements, in terms of cargo weight and passenger numbers.

$$W_{pld} = W_{cargo} + (N_{crew} + N_{pax})(W_{pax} + W_{bag}) \quad (4.2)$$

Conversely, fuel and aircraft empty weight are themselves functions of MTOW, making it necessary for an iterative calculation method to be used. Calculations are simplified by expressing these weights as fractions of MTOW. The empty weight fraction, further discussed in section 4.2.2.1, is obtained using an empirical relation based on past seaplane designs. The fuel weight fraction is found in terms of weight fractions for each mission segment.

$$W_o = \frac{W_{pld}}{1 - \frac{W_e}{W_o} - \frac{W_f}{W_o}} \quad (4.3)$$

$$\frac{W_f}{W_o} = \left(1 - \prod_{i=1}^n \frac{W_i}{W_{i-1}}\right) - \sum \Delta W_{pld} \quad (4.4)$$

,where n is the number of defined mission segments, W_i is the weight at the end of the i^{th} segment and ΔW_{pld} is the weight of payload ejected in flight.

Weight fractions for the taxi, takeoff, descent and landing segments are taken as constant, based on past experience. The range of suitable values is given by Raymer [67]. The fraction for a climb & acceleration segment is calculated using equation 4.5.

$$\left(\frac{W_i}{W_{i-1}}\right)_{CL} = \frac{f(M_i)}{f(M_{i-1})} \quad (4.5)$$

$$f(M) = \begin{cases} 0.991 - 0.007M - 0.01M^2 & \text{if } M \geq 1 \\ 1.0065 - 0.0325M & \text{if } 0.1 < M < 1 \end{cases} \quad (4.6)$$

The range of a jet aircraft can be calculated using the Breguet range equation (4.7), for steady, level flight. Rearranging, we get equation (4.8) which relates the cruise weight fraction to the cruise segment length (R), aircraft lift-to-drag ratio (L/D), engine Specific Fuel Consumption (sfc), cruise altitude and Mach number.

$$R = \frac{V_\infty}{g \cdot \text{sfc}} \frac{L}{D} \ln \left(\frac{W_1}{W_2} \right) \quad (4.7)$$

$$\ln \left(\frac{W_i}{W_{i-1}} \right) = - \frac{R \cdot g \cdot \text{sfc}}{a \cdot M \cdot (L/D)_i} \quad (4.8)$$

The weight fraction for a loiter segment is obtained by rearranging the endurance equation, also derived for steady level flight, to give eq. (4.9).

$$\ln \left(\frac{W_i}{W_{i-1}} \right) = - \frac{E \cdot g \cdot \text{sfc}}{(L/D)_i} \quad (4.9)$$

The Lift-to-Drag ratio of the aircraft has a major impact on the cruise and loiter weight fractions, however it does not remain constant with time. To account for this variation, cruise and loiter legs

may be split into smaller segments and calculated using (4.10). Based on past experience a constant L/D of 20-22 can be assumed for the first design iteration of an optimised aircraft. The Oswald efficiency (e), a measure of the wing's induced drag, can be approximated using eq. (4.11), as given by Raymer [67].

$$\left(\frac{L}{D}\right)_i = \frac{W_i/S_{ref}}{qC_{D_o} + \frac{(W_i/S_{ref})^2}{q\pi Re}} \quad (4.10)$$

$$e = 4.61 (1 - 0.045Re^{0.68}) (\cos \Lambda_{LE})^{0.15} - 3.1 \quad (4.11)$$

4.2.2.1 Empty Weight

Having estimated the fuel required to complete the design mission, the empty weight must be estimated. At this early stage in the design process, little is known of the aircraft's final configuration or dimensions, therefore the designer must rely on previous experience with similar designs. The relation between empty weight (W_e) and MTOW can be modelled using power laws (4.12).

$$W_e = 10^A \cdot W_o^B \quad (4.12)$$

The constants A and B can be determined by applying a least squares fit to weight data of existing aircraft. An aircraft database was compiled using information from volumes of Jane's "All the World's Aircraft" [1]. Aircraft blueprints or 3-view drawings were also used where possible to estimate aircraft and hull dimensions. Table 4.2 shows the power law coefficients for both flying boats and amphibians. Another set of data was also analysed, whereby and estimated weight for the undercarriage (4.13), as given by Roskam [72], was subtracted from that of amphibians.

$$W_{UC} = 62.21 \left(\frac{W_o}{1000}\right)^{0.89} \quad (4.13)$$

	A	B
Flying Boats	-0.1194	0.9744
Amphibians	-0.1558	0.9923
Both (Gear Removed)	-0.0920	0.9713

Table 4.2: Empty Weight Estimation Coefficients

The majority of aircraft used to generate this empirical relation are either general aviation aircraft or seaplanes intended for very specific missions often featuring heavy specialised equipment, weapons or armour. A correction factor of 0.8-0.9 was found to produce an initial weight estimate closer to that found by the later detailed weight breakdown calculation, thus also helping the maximum takeoff weight estimate converge over fewer iterations.

4.2.3 Thrust-to-Weight and Wing Loading

4.2.3.1 Design to Requirements

The aircraft minimum takeoff thrust (T_o) requirement, for a given reference wing area (S_{ref}), is dictated by both operational and certification requirements. Large passenger or cargo seaplanes have to abide by the FAR-25 or similar certification requirements. The major constraints are outlined below:

Cruise & Loiter: The aircraft must have sufficient thrust to maintain steady level flight at the required altitude and velocity. Therefore for the n^{th} mission segment:

$$T_o = \frac{T_o}{T_h} \frac{W_n}{L/D} \quad (4.14)$$

where T_h/T_o is the engine's thrust output at the desired altitude with respect to its output at sea level.

Rates of Climb: The ability of the aircraft to perform adequately at a set of prescribed altitudes is ensured by applying minimum rate of climb (dh/dt) constraints, which can be estimated using (4.15). For a civilian airliner or cargo transport the two critical altitudes defined are the Service and Absolute ceilings.

$$\frac{dh}{dt} = V_\infty \left[\frac{T_h/T_o}{W/W_o} \left(\frac{T}{W} \right)_o - \frac{1}{L/D} \right] \quad (4.15)$$

At the service ceiling the aircraft must be capable of climbing at a rate of 500 fpm ($2.54m/s$), while at the absolute ceiling it must be able to maintain steady level flight ($dh/dt = 0$).

Transition & Engine Failures: The FAR-25 airworthiness standards outline the minimum climb gradient (CGR) that must be achievable at different stages of takeoff, following an engine failure, or in case of an aborted landing. The thrust required for the climb gradient requirements seen in table 4.3 is given by:

$$\left(\frac{T}{W}\right)_o = \begin{cases} \frac{N_{eng}}{N_{eng}-1} \left(\frac{1}{L/D} + CGR\right) & , \text{ if one engine inoperative (OEI)} \\ \frac{1}{L/D} + CGR & , \text{ if all engines operational (AEO)} \end{cases} \quad (4.16)$$

Segment	Regulation	Number of Engines (N_{eng})		
		2	3	4
Initial climb (OEI)	FAR 25.111	0.012	0.015	0.017
Transition segment (OEI)	FAR 25.121	0.000	0.003	0.005
Second segment (OEI)	FAR 25.121	0.024	0.027	0.030
En-route (OEI)	FAR 25.121	0.012	0.015	0.017
Go-around (AEO)	FAR 25.119	0.032	0.032	0.031
Go-around (OEI)	FAR 25.121	0.021	0.024	0.027

Table 4.3: Rate of Climb requirements

The aircraft configuration, velocity and operating conditions for each of the segments referred to can be found in Roskam [69] or the relevant airworthiness directive.

Approach: A maximum wing loading (4.17) must be set in order for extremely high landing approach velocities (V_{appr}) to be avoided.

$$\left(\frac{W}{S_{ref}}\right)_{max} = \frac{\rho C_{L_{max}} \left(\frac{V_{appr}}{1.3}\right)^2}{2 \left(\frac{W_L}{W_o}\right)_{max}} \quad (4.17)$$

where $C_{L_{max}}$ is the aircraft's stall lift coefficient in the landing configuration. For seaplanes, the maximum approach velocity should not only be governed by controllability criteria but also by maximum allowable hull impact loads and the onset velocity of high speed dynamic instabilities such as skipping.

4.2.3.2 Takeoff & Landing Distance

A major sizing constraint for aircraft operating from a paved runway is the length of tarmac available. Consequently a regional airliner must be sized for much smaller runways than a super-jumbo, intended only for hub-hub operations. To comply with FAR-25 airworthiness requirements, the minimum intended operating runway length must be equal or greater to both the aircraft's normal takeoff length and its Balanced Field Length (BFL).

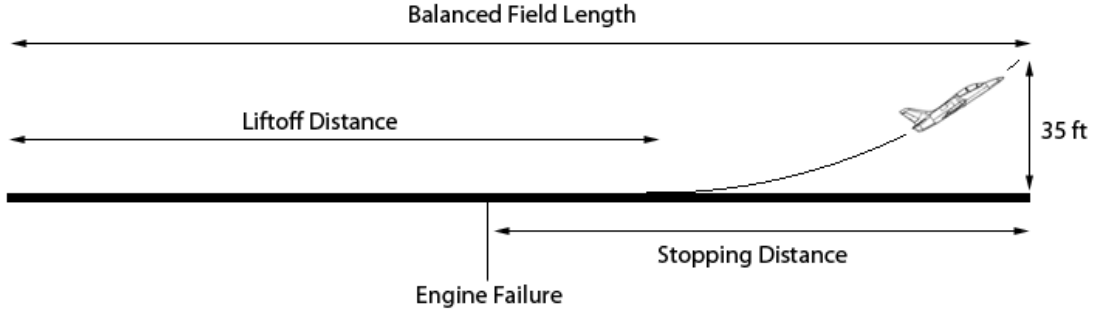


Figure 4.4: Definition of Balanced Field Length

The normal takeoff distance (S_{TO}) is defined as the distance required for an aircraft to takeoff with all engines operational and clear a 35ft high obstacle. The Balanced Field Length (shown in fig. 4.4) is the runway distance required for an aircraft, following a single engine failure, to both either continue with the takeoff and climb to the obstacle height or decelerate and come to a full stop. Roskam [69] empirically relates the BFL of aircraft to an empirical takeoff parameter (4.18). While, for the sizing of jet aircraft operating from non-paved level runways, Torenbeek [93] relates an aircraft's ground roll distance to an empirical ground friction coefficient (μ). The total takeoff distance (4.20) is obtained by adding the ground distance required for an aircraft to climb to the obstacle height (h_{OBS}).

$$BFL = \frac{0.2386 \cdot W/S_{ref}}{\sigma C_{L_{max}} T_o/W_o} \quad (4.18)$$

$$S_{TOG} = \frac{9.34 \times 10^{-4} W/S_{ref}}{\rho [C_{L_{max}} (\bar{T}/W_o - \mu) - 0.72 C_{D_o}]} \quad (4.19)$$

$$S_{TO} = S_{TOG} + \frac{h_{OBS}}{\tan \left| 0.9(\bar{T}/W_o) - 0.3/\sqrt{R} \right|} \quad (4.20)$$

where the mean thrust of a jet engine during takeoff is:

$$\bar{T}/W = k_2 \left(\frac{T}{W} \right)_o = 0.75 \frac{(5 + BPR)}{(4 + BPR)} \left(\frac{T}{W} \right)_o \quad (4.21)$$

Equations (4.19) and (4.20) can be used to obtain a crude estimate of the takeoff distance of a seaplane, using in place of the friction coefficient (μ) a representative value for a hull's hydrodynamic resistance to weight ratio. Values in the range of 0.15-0.25 were found to give reasonable values. This expression however provides only limited accuracy and does not take into account the hull shape or loading.

Initially, producing an empirical takeoff distance estimation methodology using the takeoff performance characteristics of existing aircraft was considered. However the information available was limited and not applicable if FAR-25 standards were to be met. Instead the takeoff performance of several hull shapes was estimated based on randomly selected aircraft specifications and published hull resistance data, obtained from past NACA seaplane towing tank tests. The list of the hull shapes used can be found in Appendix A and the range of values from which the aircraft aerodynamic and propulsive characteristics were chosen is given in table 4.4.

	Range		Units
C_L	0.4	1.0	
$C_{L_{max}}$	1.8	2.8	
i_w	1.0	5.0	deg
W/S_{ref}	400	7800	Nm^{-2}
$\bar{T}/W$	0.2	0.6	
$C_{\Delta o}$	0.3	1.3	
ρ	1.10	1.27	$kg \cdot m^{-3}$
W_o	4×10^5	6×10^6	N
C_{Do}	0.01	0.03	
$\mathcal{R}$	5.0	12.0	
e	0.75	0.85	
N_{eng}	2	6	

Table 4.4: Range of values used for aircraft characteristics in takeoff analysis

Force and moment data in these texts are presented using dimensionless variables, normally used in naval architecture, and which will be adopted for this report when hydrodynamic qualities are discussed. Quantities are non-dimensionalised using a boat's displacement (∇) or beam (B), the gravitational acceleration (g) and water density (w). A hull's beam is the maximum width from chine to chine and is used as the characteristic length in most high speed hydrodynamic coefficients. Therefore a hull's weight (Δ) is given by its beam loading (C_Δ).

$$C_\Delta = \frac{\Delta}{gwB^3} \quad (4.22)$$

Velocities are presented in terms of the displacement Froude number (F_{∇}).

$$F_{\nabla} = \frac{V}{\sqrt{g^{2/3}(\Delta/w)^{1/3}}} \quad (4.23)$$

At the higher velocities encountered when planing, the beam based Froude number or velocity coefficient (C_V) was found more significant and is used instead.

$$C_V = \frac{V}{\sqrt{gB}} \quad (4.24)$$

Based on the takeoff performance estimation methodology detailed by Torenbeek [93], the takeoff can be split into three distinct phases. The takeoff run, when the vessel accelerates to rotation speed (V_R), the rotation to takeoff attitude until lift-off (LOF) and the airborne phase, where the aircraft climbs to obstacle height.

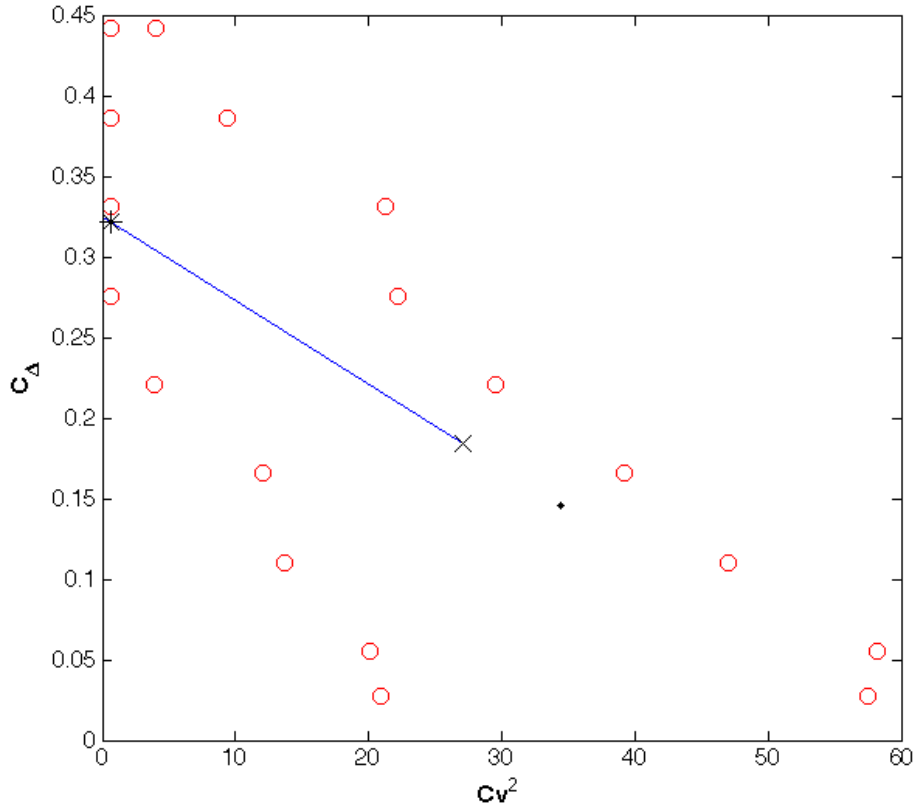


Figure 4.5: Choice of takeoff "guide" points and limits of available data for a NACA Model 11 hull Shoemaker [83]

To ensure that the entire waterborne takeoff segment lies within the bounds of the available experimental data, illustrated by red circles, two points are chosen on the Load vs. Velocity squared plane shown in figure 4.5. The first, given by the symbol $*$ represents values $(C_{V_1}^2, C_{\Delta_1})$, while the second represents $(C_{V_2}^2, C_{\Delta_2})$ and is given by the symbol $\times$. The second point further relates to the point at which rotation speed is reached, giving $C_{V_R} = C_{V_2}$. The $\diamond$ represents the calculated liftoff velocity coefficient $(C_{V_{LOF}}^2)$. Moreover, the variation of beam loading (C_{Δ}) with velocity coefficient (C_V) relates to the aircraft's lifting capacity and therefore the mean lift coefficient during the waterborne takeoff segment $(\bar{C}_L)$.

$$C_{\Delta} = C_{\Delta_1} + (C_{\Delta_2} - C_{\Delta_1}) \frac{C_V^2 - C_{V_1}^2}{C_{V_2}^2 - C_{V_1}^2} \quad (4.25)$$

$$C_{\Delta} = \frac{W_o - 0.5\rho V^2 S_{ref} \bar{C}_L}{g w B^3} = C_{\Delta_o} \left(1 - \frac{q C_L}{W/S_{ref}} \right) \quad (4.26)$$

Combining equations (4.25) and (4.26) we get:

$$C_{\Delta_o} = \frac{C_{\Delta_2} C_{V_1}^2 - C_{\Delta_1} C_{V_2}^2}{C_{V_1}^2 - C_{V_2}^2} \quad (4.27)$$

$$\bar{C}_L = \frac{2 \left(\frac{C_{\Delta_2} - C_{\Delta_1}}{C_{V_1}^2 - C_{V_2}^2} \right) C_{V_R}^2 W/S_{ref}}{\rho C_{\Delta_o} V_R^2} \quad (4.28)$$

The instantaneous acceleration experienced by the aircraft is found in terms of the aircraft's thrust, lift, drag and hydrodynamic resistance characteristics. Additional complexity is introduced by the fact that unlike the aerodynamic qualities, the hydrodynamic resistance to load ratio (R/Δ) varies and is heavily dependent upon the running attitude, C_V and C_{Δ} , as seen in figure 4.6 for a NACA Model 47 hull.

$$\frac{a}{g} = \frac{\bar{T}}{W_o} - 0.5\rho V^2 \left(\frac{C_{D_o} + \frac{C_L^2}{\pi A e} - C_L \frac{R}{\Delta}}{W/S_{ref}} \right) - \frac{R}{\Delta} \quad (4.29)$$

To obtain a dimensional value for V_R , the climb and rotation phases must be analysed. This involves an iterative procedure, whereby the velocity at liftoff (V_{LOF}) is varied between the stall speed (V_S) and the takeoff safety speed (V_2), where $V_2 \geq 1.2V_S$, until $V_2 = V_{LOF} + \Delta_2 V$.

$$V_S = \sqrt{\frac{2W/S_{ref}}{1.13\rho C_{L_{max}}}} \quad (4.30)$$

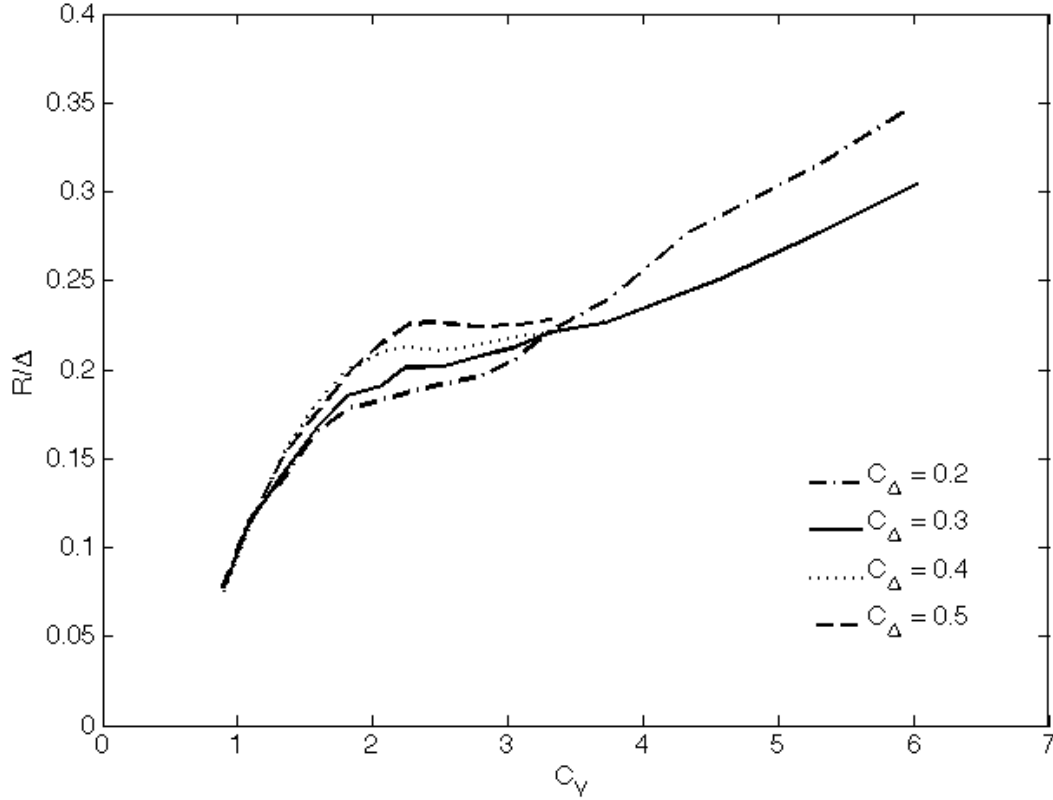


Figure 4.6: Graph of free-to-trim Resistance to Load ratio (R/Δ) vs. Velocity Coefficient (C_V) for varying beam loadings (C_Δ) for NACA Model 47 hull [99]

The airborne velocity increase ($\Delta_2 V$) is obtained, assuming a constant rate of pitch manoeuvre, using the method given by Perry [65]. This method relates the altitude gain (h) and the climb angle during transition (γ) to a non-dimensional height (F_h) and angle (F_γ) function respectively.

$$h = \frac{V_{LOF}^2}{g} \left(\frac{a}{g} \right) F_{\dot{\theta}} F_h \quad (4.31)$$

$$\gamma = \left(\frac{a}{g} \right) F_{\dot{\theta}} F_\gamma \quad (4.32)$$

where

$$F_{\dot{\theta}} = 1 + \eta \frac{g V_{LOF}}{2(a/g)} \left(\frac{d\theta}{dt} \right)_A \quad (4.33)$$

$$F_{h_{max}} = \frac{g h_{OBS}}{F_{\dot{\theta}} V_{LOF}^2 (a/g)} \quad (4.34)$$

The non-dimensional functions F_h and F_γ , which in previous texts are presented graphically, are given

as equations (4.35) and (4.36).

$$F_h = -0.1025 \frac{gt}{V_{LOF}} + 0.3701 \left(\frac{gt}{V_{LOF}} \right)^2 - 0.0008\eta + 0.0001\eta^2 + 0.0247 \frac{gt}{V_{LOF}} \eta - 0.0636 \left(\frac{gt}{V_{LOF}} \right)^2 \eta + 0.0039 \left(\frac{gt}{V_{LOF}} \right)^2 \eta^2 - 0.0017 \frac{gt}{V_{LOF}} \eta^2 \quad (4.35)$$

$$F_\gamma = 0.5769 \frac{gt}{V_{LOF}} + 0.0433 \left(\frac{gt}{V_{LOF}} \right)^2 + 0.0231\eta + 0.0031\eta^2 - 0.0359 \frac{gt}{V_{LOF}} \eta - 0.0334 \left(\frac{gt}{V_{LOF}} \right)^2 \eta + 0.0037 \left(\frac{gt}{V_{LOF}} \right)^2 \eta^2 - 0.0017 \frac{gt}{V_{LOF}} \eta^2 \quad (4.36)$$

where t is the time into the transition segment and the parameter η is given by:

$$\eta = \frac{dC_L/d\alpha}{C_{L_{LOF}}} = \frac{\pi \rho V_{LOF}^2}{W/S_{ref}} \quad (4.37)$$

The time required to reach the final climb angle of $\gamma_2 = \tan^{-1}(a/g)$, can be found by combining (4.32) and (4.36) and solving for gt/V_{LOF} . The value of F_h at time t can then be estimated using (4.35). If $F_h > F_{h_{max}}$, given in (4.34), the obstacle height has been reached before the end of the constant pitch rate manoeuvre and thus a new value for the end time t must be found solving (4.35) at $F_h = F_{h_{max}}$. Once the end time has been determined, the velocity increment and ground distance covered (S_A) during the airborne phase can be found.

$$\Delta_2 V = \left(\frac{gt}{V_{LOF}} \right) V_{LOF} \left[\frac{a}{g} - \frac{h g}{\left(\frac{gt}{V_{LOF}} \right) V_{LOF}^2} \right] \quad (4.38)$$

$$S_A = \frac{tg}{V_{LOF}} \frac{V_{LOF}^2}{g} + \frac{h_{OBS} - h}{\gamma_2} \quad (4.39)$$

If the condition $V_2 = V_{LOF} + \Delta_2 V$ is not satisfied, the value of V_2 is increased and the procedure is repeated.

Once the liftoff velocity has been found, the angle of attack at liftoff can be found as:

$$\alpha_{LOF} = \min \left(\tau_R + i_w, \frac{\frac{2W/S_{ref}}{\rho V_{LOF}^2} - C_{Lo}}{dC_L/d\alpha} \right) \quad (4.40)$$

where τ_R is the hull trim attitude at the rotation speed and i_w is the wing setting angle, referenced to the hull's keel line at the step. The rotation velocity and ground distance covered during rotation are therefore given as:

$$V_R = V_{LOF} - g \left(\frac{a}{g} \right) \frac{\alpha_{LOF} - \alpha_R}{(d\theta/dt)_R} \quad (4.41)$$

$$S_R = \frac{(V_{LOF} - V_R)(\alpha_{LOF} - \alpha_R)}{2(d\theta/dt)_R} \quad (4.42)$$

where the instantaneous acceleration includes a hydrodynamic resistance contribution, equal to 3/4 that at C_{V_R} .

The procedure detailed above must be repeated twice, once for a normal takeoff with all engines operational and once with one engine out of operation. In each case, the thrust and drag must be adjusted accordingly. As the rotation velocity is now known, the hull beam and weight can also be determined using (4.24) and (4.22).

The final part of the takeoff to be analysed is the waterborne one. The waterborne distance, for both an accelerating and decelerating vessel, is estimated by integrating the instantaneous acceleration, given by (4.29), with respect to initial and final velocities. For waterborne takeoffs, where the hydrodynamic resistance varies non-linearly with respect to velocity, numerical integration methods must be employed.

$$S_{12} = \left(\frac{1}{2g} \right) \int_{V_1}^{V_2} \frac{dV^2}{a/g} \quad (4.43)$$

For the normal takeoff case, where all engines are operational (AEO), the takeoff distance is obtained by adding the waterborne distance when accelerating from rest to the rotation velocity, to the rotation and climb phase distances.

$$S_{TO_{AEO}} = S_{0R} + S_{R_{AEO}} + S_{A_{AEO}} \quad (4.44)$$

Estimating the balanced field length requires that a single engine fail at a velocity V_x , below the rotation speed. The distances required to continue the takeoff and reach obstacle height ($S_{TO_{OEI}}$) or to abort and come to a full stop (S_{AS}) are both calculated.

$$S_{TO_{OEI}} = S_{0x} + S_{xR} + S_{R_{OEI}} + S_{A_{OEI}} \quad (4.45)$$

$$S_{AS} = S_{0x} + S_{x0} \quad (4.46)$$

The BFL is the takeoff distance at which both these distances are equal. The effect of varying failure velocity on the two takeoff distances can be seen in figure 4.7. In cases where the aircraft has a very low thrust to weight ratio, the curve for takeoff distance with one engine inoperative may never intersect with the accelerate-stop curve. In that case the BFL is taken as the worst case scenario, which is an

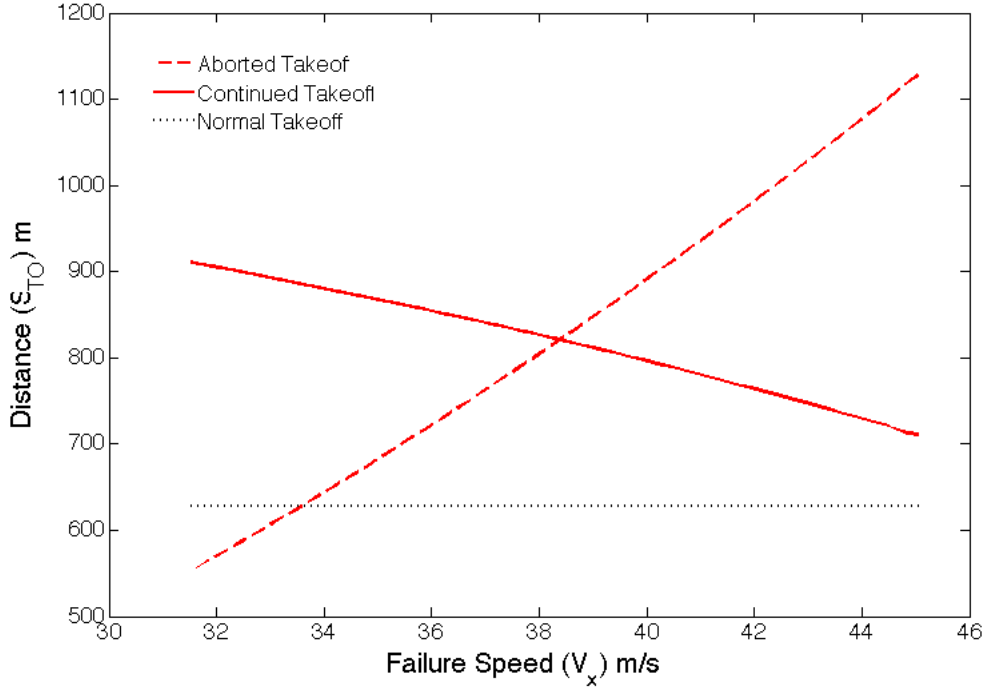


Figure 4.7: Graph of takeoff distances vs. engine failure speed for a 4 engine, NACA model 11 hull with $C_{\Delta o} = 0.326$, $W/S = 2511 N/m^2$, $\bar{T}/W = 0.378$, $W_o = 1.456 \times 10^6 N$

engine failure at rotation speed.

To facilitate and simplify the analysis described, a number of assumptions were made.

- The effects of thrust reversal, hydrodynamic braking systems, spoilers or lift dumpers are not considered.
- All engines produce equal thrust levels and are operating at maximum throttle
- The maximum thrust output per engine is assumed to be invariant with speed and equal to the mean value during takeoff given by (4.21).
- There is no wind and the water surface is perfectly calm.
- The hull's instantaneous trim angle prior to liftoff is determined solely by the hydrodynamic forces in play, as aerodynamic moments and pilot control inputs are ignored.
- A mean value of aircraft lift and parasitic drag coefficients is used despite their variation as trim/angle of attack changes during the waterborne part of the takeoff

A total of 30 random aircraft were produced and analysed for each set of hull resistance data available. The results were assembled and simplified models were generated, using multivariate linear least squares regression. The resulting formulas are based on eq. (4.19), which was deemed an appropriate physical basis.

The Balanced Field Length can be obtained using (4.47) with the values in table 4.5. The normal takeoff distance can also be found using (4.48) in conjunction with (4.20), using the constants in table 4.6.

$$\frac{W/S_{ref}}{BFL} = \rho \left\{ C_{L_{max}} \left[a_1 \frac{\bar{T}}{W} + a_2 \left(\frac{\bar{T}}{W} \right)^2 + a_3 \frac{\bar{T}}{W} \frac{N_{eng} - 1}{N_{eng}} + a_4 \frac{N_{eng} - 1}{N_{eng}} + a_5 \frac{L}{B} + \frac{a_6}{\cos \beta} \right. \right. \\ \left. \left. + a_7 \frac{N_e - 1}{N_e C_{\Delta o}} + a_8 \right] + a_9 C_{Do} \right\} + a_{10} C_{\Delta o} + a_{11} \quad (4.47)$$

$$\frac{W/S_{ref}}{S_{TOG}} = \rho C_{L_{max}} \left[b_1 \frac{\bar{T}}{W} + b_2 \left(\frac{\bar{T}}{W} \right)^2 + b_3 \frac{L_f}{B} + \frac{b_4}{\cos \beta} + b_5 C_{\Delta o} + b_6 \right] + b_7 \rho C_{Do} + b_8 \quad (4.48)$$

Equations (4.47) and (4.48) show the effect of major hull shape and size parameters on the takeoff performance of seaplanes. They are strictly valid for both single and double stepped hulls, and for specifications within the limits specified in table 4.4 and the following hull dimensions:

$$\begin{aligned} 10 \leq \beta \leq 30 & \quad 0.3 \leq C_{\Delta o} \leq 1.3 \\ 4.5 \leq L/B \leq 10.8 & \quad 2.28 \leq L_f/B \leq 5.8 \end{aligned}$$

a_1	7.15099	a_4	0.07182	a_7	0.10283	a_{10}	0.76658
a_2	-5.49267	a_5	-0.04534	a_8	-0.85773	a_{11}	-0.083248
a_3	3.07740	a_6	-0.74138	a_9	-3.088908		

Table 4.5: Constants for estimation of takeoff Balanced Field Length using eq. (4.47)

b_1	12.54183	b_3	0.08270	b_5	-0.10521	b_7	-3.73432
b_2	-6.77017	b_4	-0.90283	b_6	-1.42082	b_8	0.28393

Table 4.6: Constants for estimation of waterborne takeoff distance using eq. (4.48)

The validity of eq. (4.47) cannot be verified as the BFL is never quoted in the performance characteristics of past seaplane designs. However the general accuracy of the method used to obtain

Aircraft	Takeoff Distance (m)	
	Run	To $h_{OBS} = 50$ ft
Beriev Be-103	641	753
Canadair CL-215	761	855
Canadair CL-415MP	832	909
Gevers Genesis	352	437

Table 4.7: Takeoff performance for existing aircraft designs as predicted using eqs. (4.20) and (4.48)

these expressions can be investigated by considering the accuracy of eqs. (4.20) and (4.48) relative to available aircraft performance data, seen in table 4.8.

The takeoff run method, whose predictions can be seen in table 4.7, consistently overestimates the takeoff distance by a minor amount. This is most likely due to the fact the the empirical relations were generated neglecting the effect of aerodynamic moments, which would allow trim angles with higher or lower hydrodynamic resistance to be reached, reducing the actual BFL and water run distances. Moreover, the fact that when operating in rough water conditions, the hull's resistance is increased by upto 10%, thus affecting the takeoff distance, should be kept in mind. Nevertheless, the comparison shows that the proposed methodologies are accurate enough for initial sizing purposes.

Considering the behaviour of eq. (4.47), some insight into the way hull shape and loading parameters affect the distance can be gained. Increasing the hull's length (L/B) or forebody length (L_f/B) to beam ratios, leads to a reduction in takeoff distance. Similarly, decreasing the deadrise angle at the main step (β) or decreasing the beam loading can reduce the takeoff distance, as both lead to increases in hydrodynamic lift and a reduction in wetted area. A noteworthy aspect of waterborne takeoffs is the increased severity of having fewer engines on the BFL, particularly at lower thrust to weight ratios. The effect of the main step height and after-body keel angle on the takeoff distance were found to be statistically insignificant and should thus be overlooked during initial sizing.

No similar study was carried out for landing distances, as they were not considered as critical as the takeoff distance constraint. This rationalisation was based on the fact that, unlike land planes which have to use a combination of thrust reversal and braking systems, seaplanes will inevitably come to rest under the influence of hydrodynamic resistance forces. Therefore it is expected that if a

Aircraft	Takeoff Distance (m)		Landing Distance (m)	
	Run	To $h_{OBS} = 50$ ft	Run	From $h_{OBS} = 50$ ft
Beriev A-40	1000	1100	900	1450
Beriev Be-103	450	850	230	770
Grumman G-111	640	1356	335	670
Canadair CL-215	-	808	-	789
Canadair CL-415MP	-	814	-	665
Gevers Genesis	305	-	-	-

Table 4.8: Takeoff and Landing performance of existing seaplanes and amphibians [1]

seaplane is able to takeoff within a given distance with a reasonable amount of excess thrust, it will also be able to land in that same distance, even more so with the use of spoilers and thrust reversal. This postulation is verified by considering the performance of existing waterborne aircraft, seen in table 4.8, with the exception of the Beriev A-40.

4.2.4 The Carpet Plot

In order for the optimum combination of Thrust-to-Weight and Wing loading to be identified, curves representing the constraints specified in the previous chapters are tabulated on the same graph. For jet aircraft the curves identify the minimum required thrust level at each wing loading.

Figure 4.8 shows the carpet plot for a 350 passenger flying boat, intended to cruise for 15,500 km, with an available waterway of 3.0 km, a maximum approach speed of 150 kts and 3 engines. Based on this graph, the optimum combination of available thrust and reference wing area can be chosen, so as to minimise the maximum takeoff thrust requirement and aircraft weight. This process is also repeated for different values of stall lift coefficient for takeoff (CL_{TOmax}) and landing (CL_{Lmax}), ranging from the clean aircraft stall lift coefficient to a set of values expected to be the maximum attainable. The optimum combination is identified by minimising the performance factor:

$$f = \frac{T/W}{(W/S_{ref})^{0.84}} \left(1 + 0.03 \frac{C_{L_{TOmax}} C_{L_{Lmax}}}{(C_{L_{TOmax}})_{max} (C_{L_{Lmax}})_{max}} \right) \quad (4.49)$$

The performance factor used was derived to model the maximum level of thrust and therefore the amount of fuel the aircraft would carry. As the wing loading mostly affects the wing's structural weight, the power law given by the Class-I wing weight estimation methodology quoted by Roskam

[72] is used. The performance factor also penalises the excessive use of flaps, considering their impact on aircraft weight and system complexity. The 3% factor used for the sizing of high lift devices was determined based on attempts to redesign sample cases given by Roskam [70].

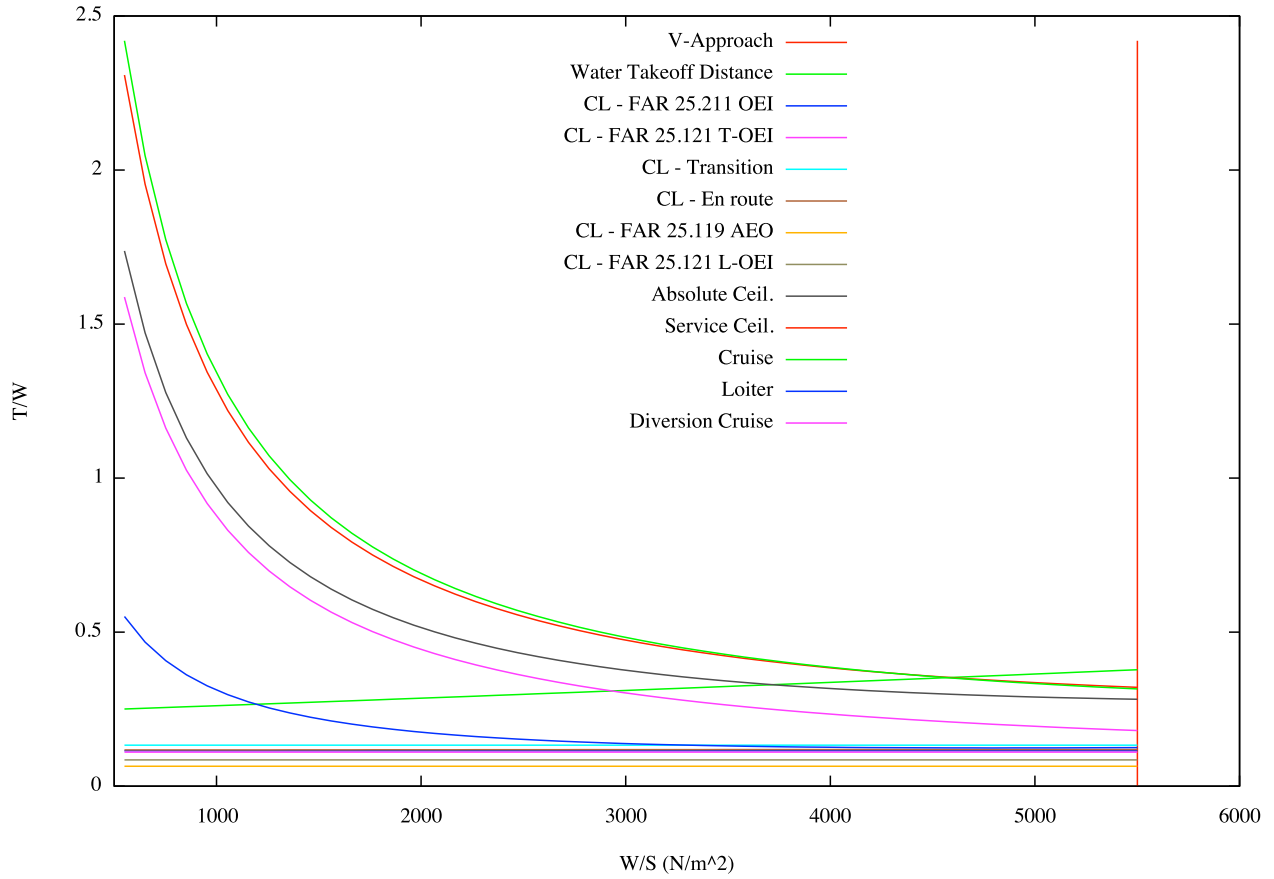


Figure 4.8: Plot of Thrust to Weight (T/W) vs. Wing Loading (W_o/S_{ref}) requirements for multiple performance constraints

Chapter 5

Geometrical Modelling

The ability to produce a wide range of individual aircraft designs, broadly based on the baseline configuration described in section 4.1.2, is paramount in order for the optimum design to be identified. Consequently the synthesis algorithm produced must be flexible enough to produce an aircraft based on distinct parameters, chosen for their ability to fully describe this particular design. This level of flexibility is particularly useful for a BWB design, as many of the subsequent aerodynamic, hydrodynamic, packaging and performance calculations (seen in figure 5.1) heavily depend on and affect the shape of the aircraft due to the iterative nature of the design process.

The number of parameters to be used ultimately depends on the complexity of the baseline design they are expected to describe and the level of control the user expects to exert. However, in order to avoid overwhelming the user, the number of parameters used must be minimised while adequate control over the output is maintained. The following sections detail the mathematical models employed to describe the aircraft's planform and cross sectional shapes, using a minimal amount of parameters.

5.1 Hull Parametrization

The shape of the hull has a major impact on a vessel's performance on water. The hull shape chosen for a particular application is often the product of an extensive tank testing program where specific points on the hull are often modified to improve its performance or address problems encountered. Such work is not possible in the preliminary design stages and therefore a parametrized hull configuration must be generated. Figure 5.2 shows a two step hull designed based on a simple set of geometrical characteristics identified by looking at past seaplane hull designs.

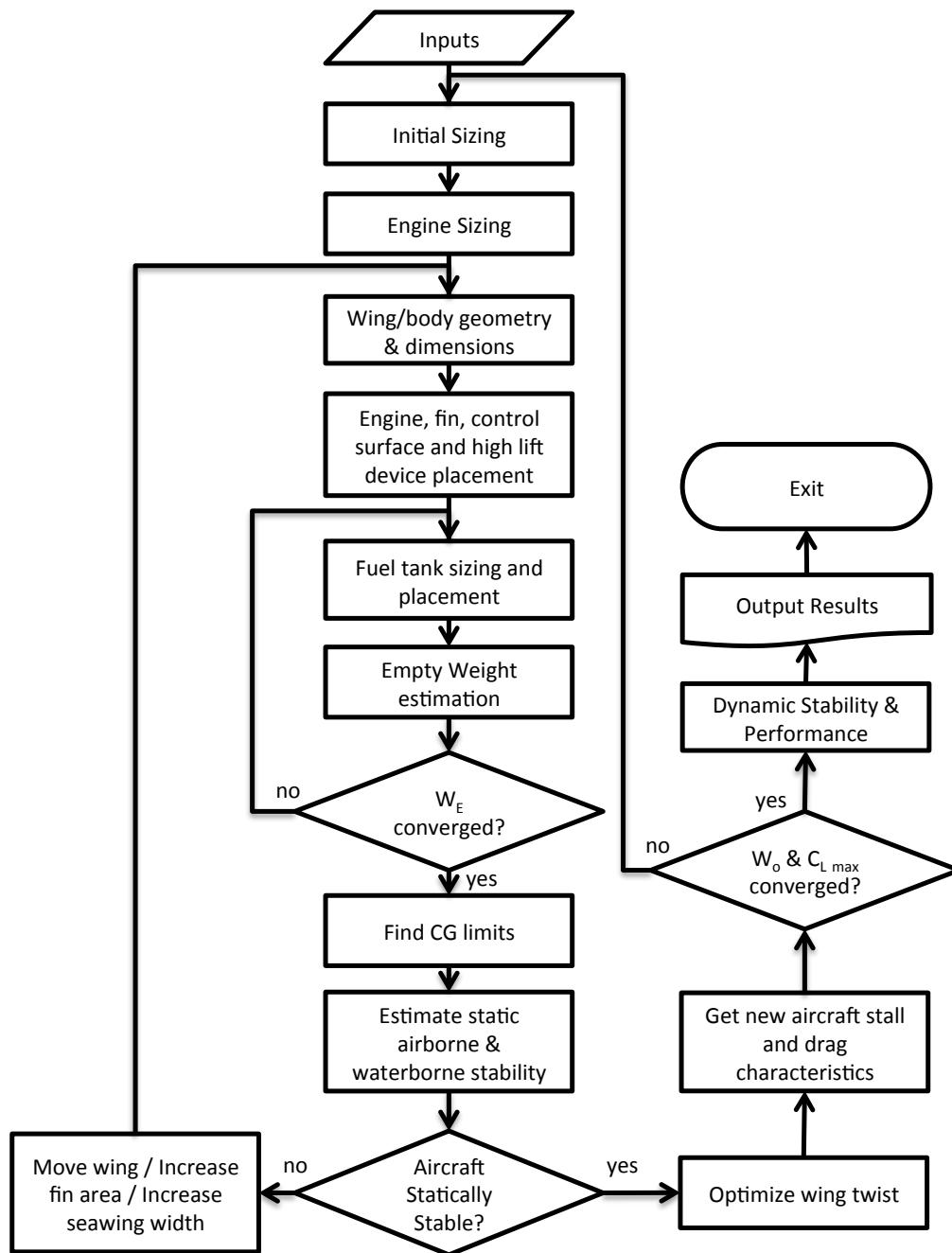


Figure 5.1: Flowchart of the design synthesis process

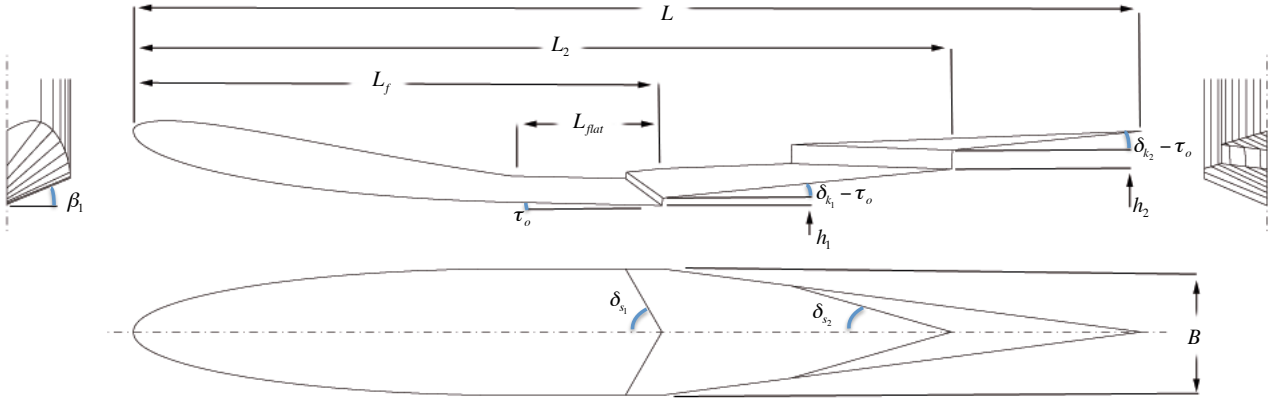


Figure 5.2: Drawing of a two step hull showing the relevant design parameters

Longitudinally a twin step hull can be split into four distinct segments. Steps are vertical discontinuities of height h on the hull, aimed at reducing afterbody suction and enabling the hull to plane at high speeds. The step planform depends on the step angle (δ_s), which describes a normal step when equal to 90 degrees, pointed steps for $\delta_s < 90\text{deg}$. and swallow steps for $\delta_s > 90\text{deg}$.

Immediately forward of the main step, hulls feature a segment of constant cross-section of length L_{flat} , approximately equal to 1 - 1.5 beams (B), aimed at improving stability when planing. The forebody flat features a constant deadrise angle (β_1), constant width equal to the beam and a flat keel. The keel of the forebody flat is often angled with respect to the reference axis by an angle τ_o and serves as the reference for all other hull keel angles.

Forward of this constant section, the remaining forebody usually features cross-sections of increasing deadrise and the hull width varies elliptically (5.1). The forebody is warped to improve rough water performance, lower the lower dynamic instability trim limits and reduce forebody impact loads. Forebody warping was found to vary substantially in past hull designs, with constant, linearly and quadratically increasing deadrise angles used. Consequently a polynomial variation of deadrise was implemented, enabling the designer to choose the optimum based on past experience. The keel height also increases nonlinearly raising the keel out of the water.

$$y = \frac{B}{2} \sqrt{1 - \left(1 - \frac{x}{L_f - L_{flat}}\right)^2} \quad (5.1)$$

The hull aft of the main step is called the afterbody. On seaplane fuselages the afterbody is streamlined in width, however here it is assumed to linearly reach zero width at $x = L$ as its blending into the wing is discussed in more detail in section 5.4.1. The keel of the afterbody sections is always flat and at an angle δ_k to the forebody keel. The afterbody keel angle ensures that the afterbody is well ventilated and out of the water at high angles of attack. The afterbody deadrise angles (β_2, β_{ab})

are also assumed constant, although examples of varying second segment and afterbody deadrise exist. The hull bottom is therefore given by:

for $0 \leq x \leq L_f - L_{flat}$

$$z_h(x, y) = z_k(x) + (L_f - x) \tan \tau_o + y \tan(\beta_1 + d\beta(L_f - L_{flat} - x)) \quad (5.2)$$

for $L_f - L_{flat} \leq x \leq L_f - y \tan(\pi/2 - \delta_{s1})$

$$z_h(x, y) = (L_f - x) \tan \tau_o + y \tan \beta_1 \quad (5.3)$$

for $L_f - y \tan(\pi/2 - \delta_{s1}) \leq x \leq L_2 - y \tan(\pi/2 - \delta_{s2})$

$$z_h(x, y) = h_1 + (x - L_f) \tan(\delta_{k1} - \tau_o) + y \tan \beta_2 \quad (5.4)$$

for $L_2 - y \tan(\pi/2 - \delta_{s2}) \leq x \leq L$

$$z_h(x, y) = h_1 + h_2 + (L_2 - L_f) \tan(\delta_{k1} - \tau_o) + (x - L_2) \tan(\delta_{k2} - \tau_o) + y \tan \beta_{ab} \quad (5.5)$$

The equations provided can be easily modified to represent a single or multiple step hull. The dimensional parameters presented above are often presented in non-dimensional form as multiples of the hull's beam.

5.2 Planform Parametrization

The choice of planform shape not only greatly affects the aerodynamic performance of an aircraft, but in the case of tailless aircraft must ensure that the required level of static stability is attained.

The planform design seen in figure 5.3 is based on the blending of a moderate to high aspect ration ($\mathcal{R}$) reference wing with a planing hull. In addition to its aspect ratio, the reference wing is defined in terms of taper ratio (λ) and quarter chord sweep ($\Lambda_{c/4}$). As the wing reference area required is already known after the initial sizing process is completed, the span (b) and root chord (c_R) for a trapezoidal reference wing are given by:

$$b = \sqrt{\mathcal{R} S_{ref}} \quad (5.6)$$

$$c_R = \frac{2}{(1 + \lambda)} \sqrt{\frac{S_{ref}}{\mathcal{R}}} \quad (5.7)$$

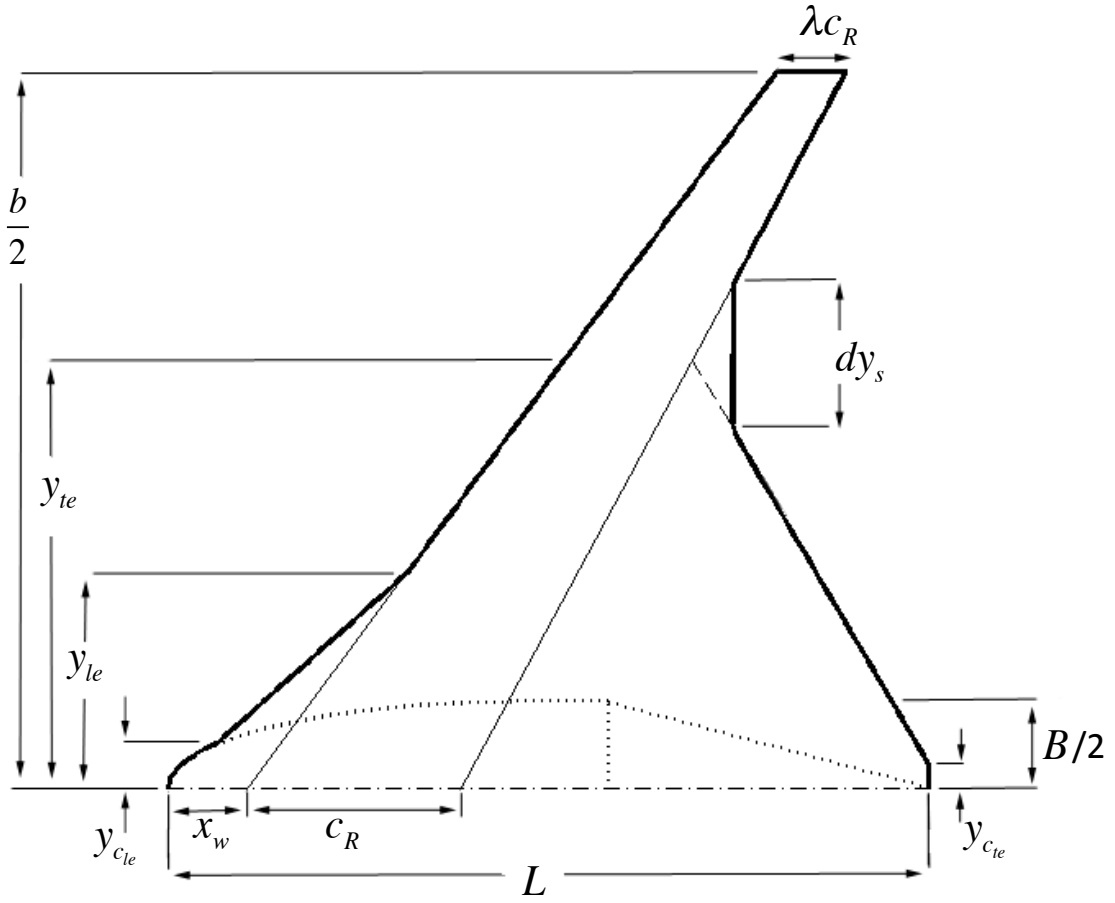


Figure 5.3: Drawing of BWB planform showing the design parameters used

$$\Lambda_{w_{le}} = \tan^{-1} \left(\tan \Lambda_{c/4} + \frac{(1-\lambda)}{\mathcal{R}(1+\lambda)} \right) \quad (5.8)$$

$$\Lambda_{wte} = \tan^{-1} \left(\tan \Lambda_{c/4} + \frac{3(\lambda - 1)}{\mathcal{R}(\lambda + 1)} \right) \quad (5.9)$$

The positioning of the reference wing along the aircraft’s length is governed by the parameter x_w , which is varied during the synthesis until the resulting planform exhibits the minimum required level of longitudinal static stability.

When blended with the wing, a portion of the hull's forward section retains its elliptical shape, given by equation (5.10), in order for the required level of forebody warp to be maintained and to house the cockpit. The rest of the wing leading edge ($y > y_{cle}$) is governed by the parameter y_{le} which dictates how far along the semi-span, the cabin leading edge meets the reference wing.

$$x_{le} = (L_f - L_{flat}) \left(1 - \sqrt{1 - \frac{y^2}{0.25B^2}} \right) \quad (5.10)$$

$$\Lambda_{le} = \begin{cases} \tan^{-1} \left(\frac{x_w + y_{le} \tan \Lambda_{w_{le}} - (L_f - L_{flat}) \left(1 - \sqrt{1 - \frac{y_{c_{le}}^2}{0.25B^2}} \right)}{y_{le} - y_{c_{le}}} \right) & y_{c_{te}} \leq y \leq y_{le} \\ \Lambda_{w_{le}} & y_{le} \leq y \leq b/2 \end{cases} \quad (5.11)$$

The trailing edge is described in much the same way. For $y < y_{c_{the}}$ the trailing edge sweep is zero. Thereafter the sweep is such that the trailing edge meets the reference wing trailing edge at $y = y_{te}$, at which point it follows the reference wing. In order for very abrupt changes in sweep to be avoided a straight section can also be added about this point. The variation in leading and trailing edge sweep is given by equations 5.11 and 5.12 respectively.

$$\Lambda_{te} = \begin{cases} 0 & 0 \leq y \leq y_{c_{te}} \\ \tan^{-1} \left(\frac{x_w + c_R + (y_{te} - 0.5dy_s) \tan \Lambda_{w_{te}} - y_{c_{te}}}{y_{te} - 0.5dy_s - y_{c_{te}}} \right) & y_{c_{te}} \leq y \leq y_{te} - dy_s/2 \\ 0 & y_{te} - dy_s/2 \leq y \leq y_{te} + dy_s/2 \\ \Lambda_{w_{te}} & y_{te} + dy_s/2 \leq y \leq b/2 \end{cases} \quad (5.12)$$

In order for the parameters y_{le} , y_{te} and dy_s to vary with the aircraft's span, within the synthesis they are expressed as invariant fractions of the semi-span. Similarly, the hull based parameters $y_{c_{le}}$ and $y_{c_{the}}$ are expressed in terms of half the hull's beam and the reference wing position (x_w) in term of the aircraft length (L).

So as to avoid the creation of unfeasible planforms, the previously specified values are altered to conform to the following of minimum and maximum sweep angles.

$$5 \text{ deg.} \leq \Lambda_{le} \leq 75 \text{ deg.}$$

$$-70 \text{ deg.} \leq \Lambda_{te} \leq -5 \text{ deg.}$$

5.3 Airfoil Parametrization

The shape of an airfoil is responsible for the majority of lifting characteristics of a wing. Specific airfoil coordinates can be used to generate a wing, however when various parameters such as thickness to chord ratio and trailing edge position vary along the span of a wing it is advantageous if airfoil shapes could be described using a reduced number of parameters. Conformal mappings are

often used to generate airfoil shapes however their use in conjunction with simple geometric parameters is rather complex. Instead a variation on the PARSEC parametric airfoil generation method, originally presented by Sobieczky [87], was opted for. This method has been previously used in wing and airfoil optimisation problems, such as the optimisation of transonic airfoils or wind turbine blades.

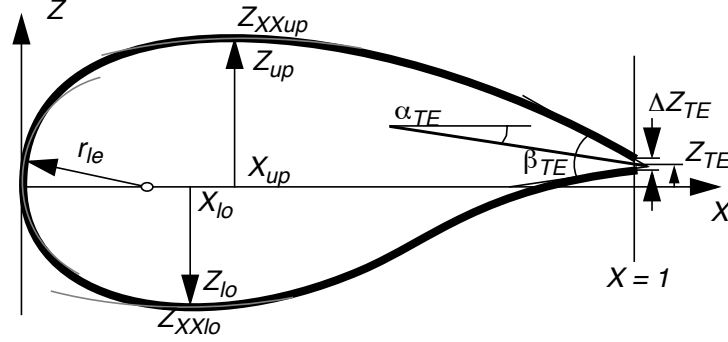


Figure 5.4: Sample PARSEC airfoil displaying the governing geometric parameters [87]

As seen in figure 5.4 the airfoil shape is defined in terms of the airfoil's geometric characteristics at four distinct points. The leading edge is described in terms of the upper (R_u) and lower (R_l) leading edge radii. The trailing edge is defined in terms of the trailing edge height (Z_{te}), trailing edge thickness (ΔZ_{te}) and the trailing edge setting (α_{te}) and inside (β_{te}) angles. The rest of the airfoil shape is defined by the position of the upper curve maximum (X_u, Z_u) and lower curve minimum (X_l, Z_l), as well as the curvatures Z_{XX_u} and Z_{XX_l} at those points. Based on these parameters the upper and lower airfoil curves can each be represented by an n^{th} order polynomial of the form:

$$z(x) = \sum_{i=1}^n a_i x^{(2i-1)/2} \quad (5.13)$$

The original methodology involved the use of a 6^{th} order polynomial, however a polynomial of such high order was found to be rather susceptible to Runge's phenomenon, often producing rather irregular airfoil shapes. Instead, the use of a 4^{th} order polynomial forward of the minimum or maximum point and a 5^{th} order polynomial aft proved to produce more conventional airfoils. The constants a_i in (5.13) can be found by solving the following systems of simultaneous equations, derived by matching the curves to the geometric requirements at each point.

Upper curve, $0 \leq x \leq X_u$

$$\begin{aligned}
a_1 &= \sqrt{2R_u} \\
\sum_{i=1}^4 a_i \sqrt{X_u}^{(2i-1)} &= Z_u \\
\sum_{i=1}^4 a_i (2i-1) \sqrt{X_u}^{(2i-2)} &= 0 \\
\sum_{i=1}^4 a_i (2i-1)(2i-2) \sqrt{X_u}^{(2i-3)} &= 4X_u Z_{XX_u}
\end{aligned} \tag{5.14}$$

Upper curve, $X_u \leq x \leq 1$

$$\begin{aligned}
\sum_{i=1}^5 a_i &= Z_{te} + 0.5\Delta Z_{te} \\
\sum_{i=1}^5 a_i (2i-1) &= -2 \tan(\beta_{te}/2 + \alpha_{te}) \\
\sum_{i=1}^5 a_i \sqrt{X_u}^{(2i-1)} &= Z_u \\
\sum_{i=1}^5 a_i (2i-1) \sqrt{X_u}^{(2i-2)} &= 0 \\
\sum_{i=1}^5 a_i (2i-1)(2i-2) \sqrt{X_u}^{(2i-3)} &= 4X_u Z_{XX_u}
\end{aligned} \tag{5.15}$$

Lower curve, $0 \leq x \leq X_l$

$$\begin{aligned}
a_1 &= -\sqrt{2R_l} \\
\sum_{i=1}^4 a_i \sqrt{X_l}^{(2i-1)} &= Z_l \\
\sum_{i=1}^4 a_i (2i-1) \sqrt{X_l}^{(2i-2)} &= 0 \\
\sum_{i=1}^4 a_i (2i-1)(2i-2) \sqrt{X_l}^{(2i-3)} &= 4X_l Z_{XX_l}
\end{aligned} \tag{5.16}$$

Lower curve, $X_l \leq x \leq 1$

$$\begin{aligned}
\sum_{i=1}^5 a_i &= Z_{te} - 0.5\Delta Z_{te} \\
\sum_{i=1}^5 a_i (2i-1) &= 2 \tan(\beta_{te}/2 - \alpha_{te}) \\
\sum_{i=1}^5 a_i \sqrt{X_l}^{(2i-1)} &= Z_l \\
\sum_{i=1}^5 a_i (2i-1) \sqrt{X_l}^{(2i-2)} &= 0 \\
\sum_{i=1}^5 a_i (2i-1)(2i-2) \sqrt{X_l}^{(2i-3)} &= 4X_l Z_{XX_l}
\end{aligned} \tag{5.17}$$

Figure 5.5 compares a NASA SC(2)-0610 airfoil with the closest airfoil shape that can be generated using the parametric airfoil method discussed using the parameters in table 5.1.

Occasionally packaging constraints will require that the upper airfoil surface be reshaped to accommodate a larger cabin or cargo bay. When such situations are encountered, the upper curve is

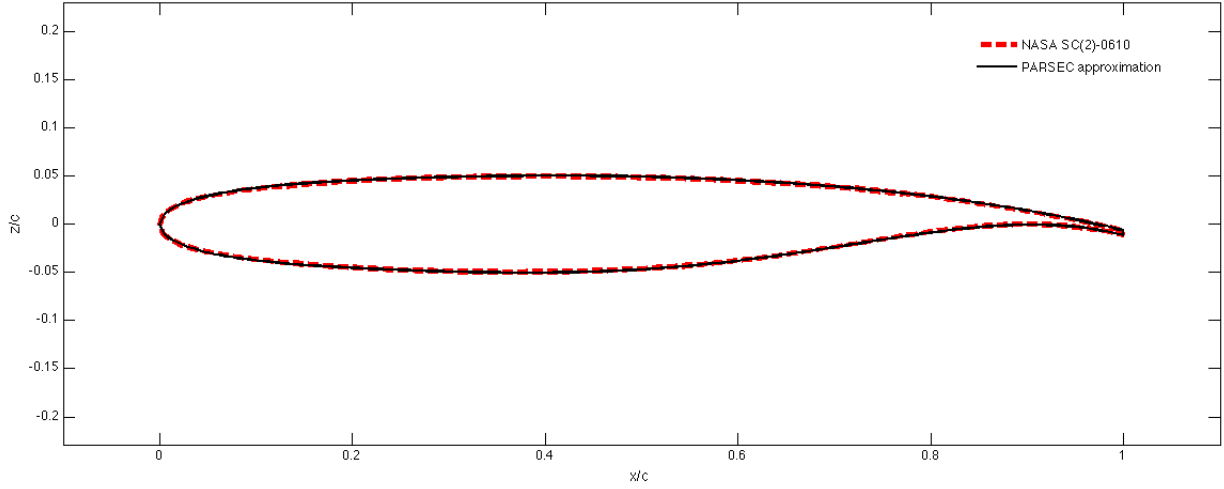


Figure 5.5: Comparison of a NASA SC(2)-0610 airfoil and its parametrized equivalent

R_u	0.01031	R_l	0.01061	Z_{te}	-0.00915
X_u	0.41318	X_l	0.38238	ΔZ_{te}	0.00490
Z_u	0.05005	Z_l	-0.04995	α_{te}	14.20566 deg.
Z_{XX_u}	-0.27216	Z_{XX_l}	0.39578	β_{te}	-1.51831 deg.

Table 5.1: Geometric parameters for a NASA SC(2)-0610 airfoil

instead modelled using three polynomials. The exact order of polynomials depends on whether the forward, aft or both sections must be modified and is discussed further in Appendix B.

5.4 3D Modelling

Generating a three dimensional representation of the aircraft is a necessary step both in evaluating the aircraft's aerodynamic characteristic and ensuring that the packaging requirements discussed in chapter 6 are met. Given the similarity between the baseline configuration and a standalone wing, modelling the wing-body combination by blending a series of airfoil like, spanwise cross-sections was considered optimum.

For the purposes of simplifying the aircraft shape generation, the aircraft can be split into three distinct spanwise components. The hull section extends from the centreline for approximately half a beam length. This section forms the aircraft's planing hull which is used to facilitate waterborne operation. Immediately outboard is the seawing, a section intended to house the remainder of the passenger cabin and improve waterborne lateral stability at low speed. Finally the outer wing section extends outboard to the wingtips.

5.4.1 Hull Section & Seawing

The hull section seamlessly blends a three-dimensional hull, designed for optimum high speed planing, with an aerodynamically favourable airfoil cross section. The section's design is heavily dependent on the desired planform, discussed in section 5.2, hull dimensions and the centreline thickness to chord ratio $((t/c)_o)$ required to fulfil packaging criteria.

The airfoil sections are generated using a combination of eqs. (5.2) to (5.5) and eq. (5.13). The hull shape is maintained for $x'_f \leq x' \leq x'_a$, where x' is the longitudinal ordinate nondimensionalized by the local chord length, while forward and aft of that section airfoil curves are generated. The coefficients for these airfoil curves are obtained by solving the following systems of simultaneous equations:

for $0 \leq x' \leq x'_f$

$$\begin{aligned} a_1 &= -\sqrt{2R} \\ \sum_{i=1}^3 a_i \sqrt{x'_f}^{(2i-1)} &= z'_f \\ \sum_{i=1}^3 a_i (2i-1) \sqrt{x'_f}^{(2i-2)} &= -2\sqrt{x'_f} \tan \delta_f \end{aligned} \quad (5.18)$$

for $x'_a \leq x' \leq 1$

$$\begin{aligned} \sum_{i=1}^4 a_i &= Z_{te} - 0.5\Delta Z_{te} \\ \sum_{i=1}^4 a_i (2i-1) &= 2 \tan (\beta_{te}/2 - \alpha_{te}) \\ \sum_{i=1}^4 a_i \sqrt{x'_a}^{(2i-1)} &= z'_a \\ \sum_{i=1}^4 a_i (2i-1) \sqrt{x'_a}^{(2i-2)} &= 2\sqrt{x'_a} \tan \delta_a \end{aligned} \quad (5.19)$$

The cross-section chosen for the aircraft's centreline has a major impact on the remaining section. To enable the use of the previously defined airfoil shape parameters for varying airfoil thicknesses, the height parameters Z'_u , Z'_l and Z'_{te} are defined as fractions of thickness to chord ratio. The leading edge radii R_u and R_l are also treated as functions of thickness using an n^{th} order polynomial. Also defined is the hull cut-off point (X_{co}) , such that at the centreline:

$$\begin{aligned} x'_f &= \frac{L_f/B - L_{flat}/B}{L/B} \\ z'_f &= Z'_l \left(\frac{t}{c} \right)_o + \frac{L_f/B - L_{flat}/B}{L/B} \tan \tau_o \\ \delta_f &= \tau_o \end{aligned}$$

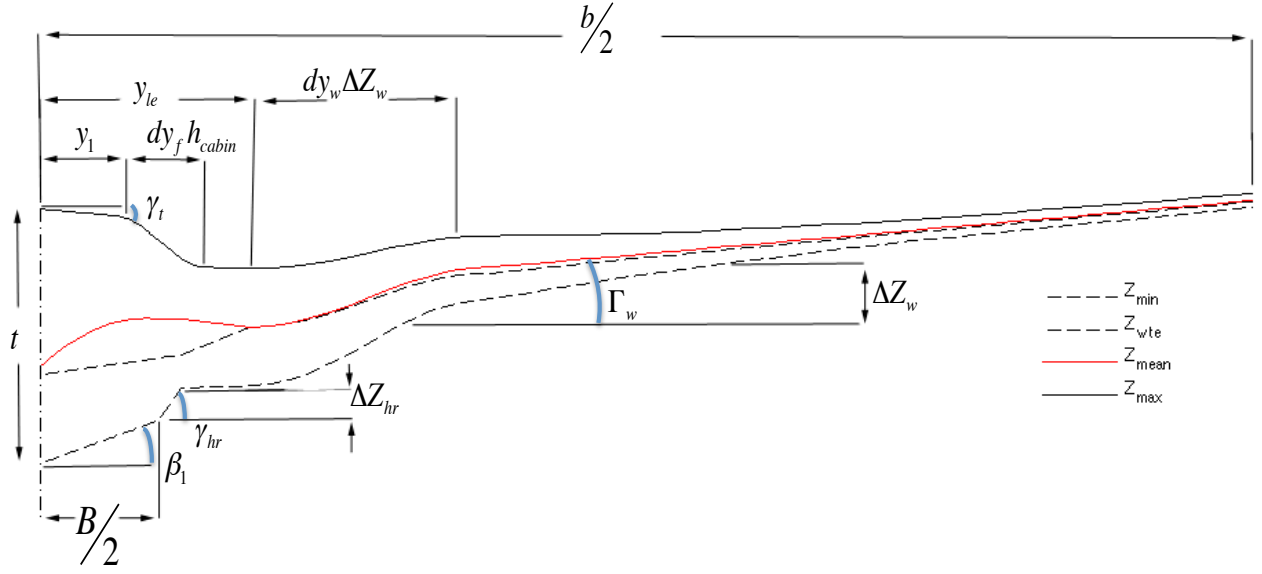


Figure 5.6: Plot of the spanwise variation of the major wing vertical geometry descriptors

$$\begin{aligned}
 x'_a &= X_{co} \\
 z'_a &= Z'_l \left(\frac{t}{c} \right)_o + \frac{z_h(x'_a L, 0)}{L} \\
 \delta_a &= \delta_k(x = x'_a L) \\
 Z_{te} &= Z'_{te}(t/c)_o
 \end{aligned}$$

If the value of X_{co} is higher than 0.9 or the trailing edge height $(Z'_{te}(t/c)_o - \Delta Z_{te}/2)$ is lower than that of the keel of the last hull section if extended to the trailing edge, the predefined parameters governing the trailing edge geometry are overridden such that:

$$\begin{aligned}
 x'_a &= 1 \\
 \alpha_{te} &= \beta_{te}/2 - \delta_k|_{x=L} + \tau_o \\
 Z'_{te} &= Z'_l + \left(\frac{z_h(L, 0)}{L} + \frac{\Delta Z_{te}}{2} \right) / \left(\frac{t}{c} \right)_o
 \end{aligned}$$

Moving further outboard from the centreline, the sectional thickness to chord ratio changes due to the hull's deadrise angle and shaping requirements for the top wing surface and can thus be expressed as:

$$\frac{t}{c} = \frac{Z_{max} - Z_{min}}{x_{te} - x_{le}}$$

where the airfoil maxima (Z_{max}) and minima (Z_{min}) can be seen in figure 5.6 and are given by:

$$Z_{max}(y) = Z'_u(t/c)_o L + y \tan \gamma_t - dZ_f(y) \quad (5.20)$$

$$Z_{min}(y) = \begin{cases} Z'_l(t/c)_o L + y \tan \beta_1 & \text{if } y \leq B/2 \\ Z'_l(t/c)_o L + 0.5B \tan \beta_1 + B\Delta Z_{hr} & \text{if } y = y_{hr} \end{cases} \quad (5.21)$$

As seen in figure 5.6, in order to maintain the step geometry and the separation of water from the hull's chines during planing, at $y = B/2$ the hull rises abruptly by a predefined rise to beam ratio (ΔZ_{hr}). To ease the aerodynamic impact of such a discontinuity the rise occurs at an angle γ_{hr} to the horizontal, usually of the order of 65 deg. Therefore the semi-width of the entire hull section is:

$$y_{hr} = B(0.5 + \Delta Z_{hr} / \tan \gamma_{hr})$$

One should note that the hull rise angle (γ_{hr}) must be sufficient for the water flow to separate at the chines when planing. In cases where the use of multiple cabin floors is necessary but an upper floor does not span as wide as those below it, the top surface follows the cabin contours, reducing the local sectional thickness by dZ_f and enabling the placement of emergency exits on the sides of the cabin.

$$dZ_f(y) = \begin{cases} 0 & \text{if } y < y_1 \\ h_{cabin} \sin^2 \left(\frac{\pi}{2} \frac{y-y_1}{dy_f h_{cabin}} \right) & \text{if } y_1 \leq y \leq y_1 + dy_f h_{cabin} \\ h_{cabin} & \text{if } y > y_1 + dy_f h_{cabin} \end{cases} \quad (5.22)$$

Equation (5.22), where y_1 is the point of cabin height change, h_{cabin} is the cabin height and dy_f is the fairing width to height ratio, can be used concurrently in case of multiple cabin height changes. To avoid issues in blending with the outer wing section a minimum value of Z_{max} is enforced, dependent on the outer wing's vertical placement.

Also shown in figure 5.6 are the leading edge (Z_{mean}) and trailing edge (Z_{wte}) height variations. To maintain the required level of forebody warp and a smooth airfoil shape, the airfoil leading edge height must vary with span. Therefore if at the centreline $Z_{mean} = 0$ and $z_h(x, y)$ is the hull bottom height:

$$Z_{mean}(y) = \frac{z_h(x_{le}, y)}{2} + \left(\frac{Z_{max}(y_{hr}) + Z_{min}(y_{hr})}{2} - \frac{z_h(x_{le}, y)}{2} \right) \frac{y}{y_{hr}} \quad (5.23)$$

The wing trailing edge follows the afterbody deadrise to keep the trailing edge out of the water at high angle of attack.

$$Z_{wte}(y) = Z'_{te}(t/c)_o L + y \tan(f_{te} \cdot \beta_{ab}), \quad 0 < f_{te} < 1 \quad (5.24)$$

Based on the above vertical shape constraints the individual unit chord airfoil section for $0 < y \leq B/2$ can be generated using the following altered parameters to get the forward and aft sectional shapes.

$$\begin{aligned}
Z_u &= \frac{Z_{max}(y) - Z_{mean}(y)}{x_{te} - x_{le}} \\
R_u &= f(2Z_u) \\
R_l &= f\left(2\frac{Z_{mean}(y) - Z_{min}(y)}{x_{te} - x_{le}}\right) \\
x'_f &= \frac{(L_f - L_{flat})\left(1 - \sqrt{1 - \frac{4y^2}{B^2}}\right) - x_{le}}{x_{te} - x_{le}} \quad 0.1 \leq x'_f \leq 0.2 \\
z'_f &= \frac{z_h(x_{le} + x'_f(x_{te} - x_{le}), y) - Z_{mean}(y)}{x_{te} - x_{le}} \\
\delta_f &= \tan^{-1}\left(\frac{z_h(x_{le} + x'_f(x_{te} - x_{le}), y) - z_h(x_{le} + x'_f(x_{te} - x_{le}) + dx, y)}{dx}\right) \\
x'_a &= \min\left(\frac{X_{co}L - x_{le}}{x_{te} - x_{le}}, \frac{L - 2y(L/B - L_f/B) - x_{le}}{x_{te} - x_{le}}\right) \\
z'_a &= \frac{z_h(x_{le} + x'_a(x_{te} - x_{le}), y) - Z_{mean}(y)}{x_{te} - x_{le}} \\
\delta_a &= \tan^{-1}\left(\frac{z_h(x_{le} + x'_a(x_{te} - x_{le}) + dx, y) - z_h(x_{le} + x'_a(x_{te} - x_{le}), y)}{dx}\right) \\
z'_{te} &= \frac{Z_{wte}(y) - Z_{mean}(y)}{x_{te} - x_{le}}
\end{aligned}$$

For $x'_f \leq (x - x_{le})/(x_{te} - x_{le}) \leq x'_a$ the original hull shape is maintained and can be obtained using:

$$z = Z_{min}(y) + z_h(x, y) \quad (5.25)$$

where z_h is given by eqs. (5.3) to (5.5). Finally at $y = y_{hr}$ the airfoil is simply generated using the originally supplied airfoil parameters for the lower curve.

The sawing section extends from the end of the hull (y_{hr}) to y_{le} . The cross sections of this part are pure airfoils obtained using the methods of section 5.3. As for the hull, the sectional height parameters used are fractions of the local thickness to chord ratio. The local airfoil maxima and minima are again defined in terms of upper and lower angles of dihedral. The top surface is generated in the same way as for the hull section. The bottom surface is linearly interpolated between y_{hr} and y_{sw} , where $y_{sw} \leq y_{le}$, and a cross section of constant characteristics is used thereafter.

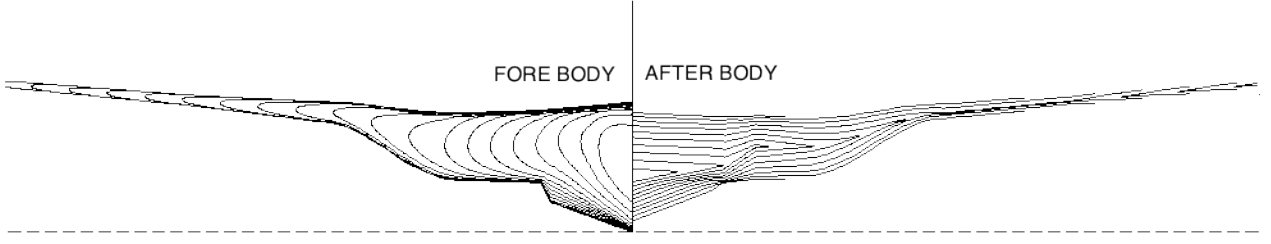


Figure 5.7: Chordwise hull cross-sections for a seaplane with a blended single step hull

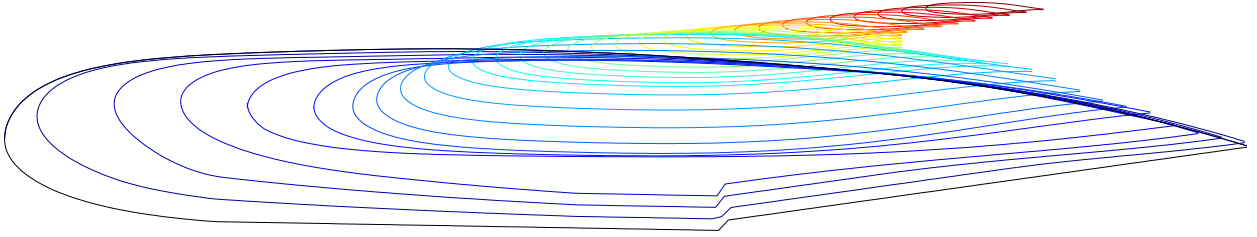


Figure 5.8: Spanwise airfoil cross-sections for a seaplane with a blended single step hull

The resulting chordwise and spanwise cross-sections, seen in figures 5.7 and 5.8, show that the hull can be efficiently blended with airfoil sections without compromising the hull shape. The only visible side effect on the resulting hull lines is the addition of moderate chine flaring on some forebody sections, which although unintentional reflects current hull design practices.

5.4.2 Outer Wing Section

The outer wing section features a wing made of predefined airfoil sections, which is blended with the seawing. In order to keep the wingtips clear of the water when at a roll angle, the wing is high mounted, such that the leading edge height Z_{mean} is increased by ΔZ_w over a span equal to $dy_w \Delta Z_w$. The blending section is generated by interpolating the two sets of airfoil coordinates using a trigonometric function.

$$\begin{aligned} z_i(y) &= z_i(y_{le}) + 0.5(z_i(y_{le} + dy_w \Delta Z_w) - z_i(y_{le})) \left(1 - \cos\left(\pi \frac{y - y_{le}}{dy_w \Delta Z_w}\right)\right) \\ x_i(y) &= x_i(y_{le}) + 0.5(x_i(y_{le} + dy_w \Delta Z_w) - x_i(y_{le})) \left(1 - \cos\left(\pi \frac{y - y_{le}}{dy_w \Delta Z_w}\right)\right) \end{aligned} \quad (5.26)$$

Use of eq. (5.26) ensures a gentle transition, not penalising aerodynamic performance and in most cases provides sufficient side surface area for the placement of emergency exits on the cabin sides.

For $y_{le} + dy_w \Delta Z_w \leq y \leq b/2$ creating the wing is extremely straightforward, using the methods in

section 5.3. The airfoil sections used are fully defined by the user, including their thickness to chord ratios. Multiple airfoil sections can be linearly blended over specified span fractions of the outer wing, with each section capable of featuring a distinct wing dihedral angle (Γ). The resulting airfoil curves are given by:

$$z(x, y) = Z_{mean}(y) + (x_{te} - x_{le}) \sum_{i=1}^n a_i(y) x^{(2i-1)/2} \quad (5.27)$$

To improve the accuracy of the airfoil coordinates generated in the leading and trailing edges when using only a limited number of points, trigonometric coordinate distributions were used.

Chapter 6

Systems Packaging

Having completed the initial design process and obtained an initial estimate of the wing-body shell and its dimensions, the aircraft's ability to house all the necessary systems must be evaluated. The placement of the various systems follows the general locations laid out in the baseline configuration, yet must be flexible enough to mirror the innate flexibility of the aircraft's geometry.

It should be noted that the packaging requirements not only affect the vehicle's weight and balance but the shell geometry as well, ultimately having an impact on the aircraft's aerodynamic performance. Consequently the three-dimensional geometry must be obtained in conjunction with updated packaging constraints in an iterative fashion.

6.1 Pressurised hull sizing

The most critical packaging constraint for a BWB passenger or cargo aircraft is the need to accommodate the required number of passengers and associated cargo volume. The methods described in this sections are used to fill a previously obtained structural shell with the minimum number of pressurised passenger and cargo bay tubes necessary to accommodate a predefined number of passengers in an all economy configuration and their baggage. The effect these packaging constraints have on the aircraft's centreline thickness-to-chord ratio, and how it is calculated, is subsequently discussed.

6.1.1 Passenger Cabin

As seen in figure 6.1 the cabin is made up of several conjoint slender passenger cabins, seating m_{pax} passengers per row. Each bay features a door for each aisle, placed along the wing leading edge. All these doors are connected by a single aisle running along the wing's leading edge, forward of the

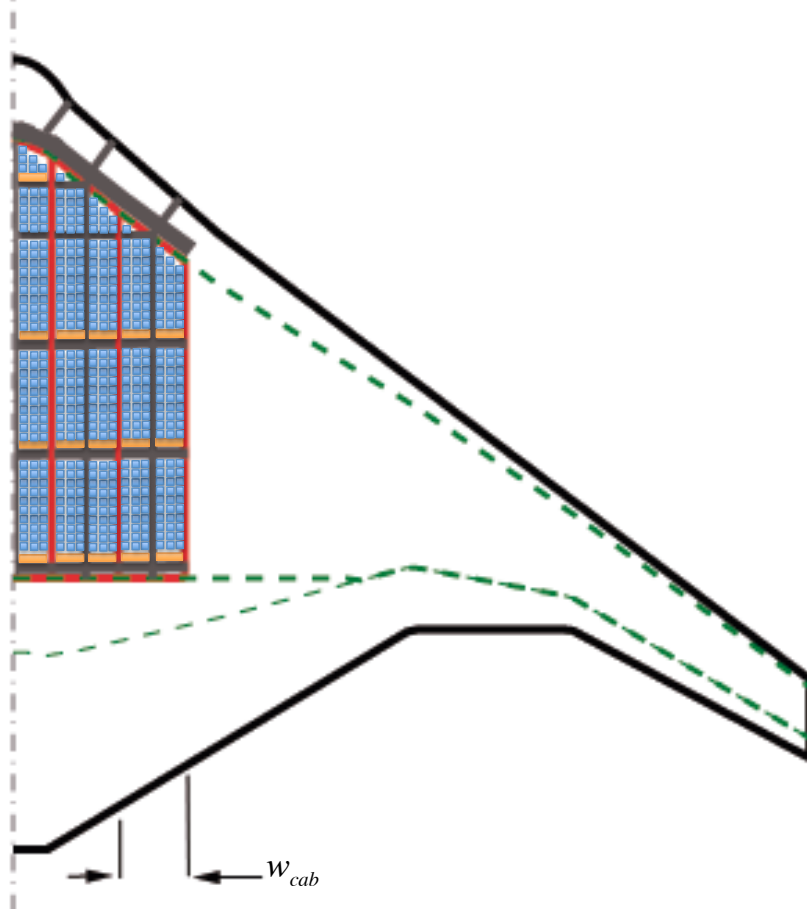


Figure 6.1: View of the passenger cabin configuration showing front and rear spars (green), cabin walls (red), aisles (grey), seats (blue) and service areas (orange)

cabin. To abide by the requirement that no more than four-abreast seating be used, the number of aisles necessary is:

$$N_{aisle} = \text{ceiling} \left(\frac{m_{pax} - 6}{4} \right) + 1 \quad (6.1)$$

Therefore the width of each cabin section (w_{cab}) is:

$$w_{cab} = t_{wall} + m_{pax}w_{seat} + N_{aisle}w_{aisle} + (m_{pax} + N_{aisle} + 1)w_{arm} \quad (6.2)$$

where t_{wall} is the thickness of the dividing walls, w_{seat} and w_{arm} are the seat cushion and armrest widths respectively and w_{aisle} is the aisle width. Another feature of the baseline cabin are cross-aisles, which allow the movement of passengers between cabin bays and towards possible exits. Each leading edge door is linked to its counterpart on the aircraft's opposite side with a cross aisle. Additional cross-aisles are situated in the aft, rectangular, portion of the cabin, spaced N_{row} rows apart, allowing access to the seaplane's side emergency exits.

When sizing the cabin, floor area per person must also be allocated to lavatories (A_{lav}), galleys (A_{galley}). Staircases are only used when multiple floors are needed. In addition to mirrored, 35 degree pitch, straight staircases at the front of the cabin, spiral staircases are situated at the back of each cabin bay.

$$A_{stair} = 2.85 w_{stair} h_{cabin} + 1.02 N_{bay} \frac{h_{cabin}^2}{\pi}$$

Therefore the number of passengers that can be accommodated for a given seat pitch (p) can be found by dividing the available cabin floor area by the area required per passenger.

$$N = \frac{A_{floor} - A_{aisle} - A_{c-aisle} - A_{stair}}{p(w_{seat} + w_{arm}(m_{pax} + N_{aisle} + 1)/m_{pax}) + A_{lav} + V_{galley}/h_{cabin}} \quad (6.3)$$

p	=	1.00 m	w_{seat}	=	0.432 m
w_{arm}	=	0.064 m	w_{aisle}	=	0.534 m
$w_{c-aisle}$	=	1.00 m	A_{lav}	=	0.02 m ² /pax
V_{galley}	=	0.03 m ³ /pax	h_{cabin}	=	2.500 m
N_{row}	=	10	w_{stair}	=	1.500 m

Table 6.1: Cabin sizing parameters for a typical all economy class cabin. [93] [71]

If the use of multiple floors is necessary, a weight can be applied to the division of passengers between floors, ensuring that the width of the upper floors is less than those below. This is beneficial as the resulting wing shape allows the placement of emergency exits on the cabin sides. Upper floors follow the layout of those below so that the structural ribs housed within the dividing walls remain uninterrupted from the bottom to the top of the structure.

If a single central bay is required to produce the number of seats closest to the design number of passengers, and the predefined number of seats abreast produces a non-symmetrical seat layout, m_{pax} is increased.

6.1.2 Cargo bays

Cargo can be placed either outboard of the passenger cabins or within the seaplane's hull, underneath the cabin. In modern airliners baggage and loose cargo is first stored in Unit Load Devices,

which enable the handling and stowage of a standardised unit. The cargo bay sizing described below is based on specific, user defined container dimensions. The total number of ULDs necessary to store all cargo volume is therefore:

$$N_{ULD} = \text{ceiling} \left(\left(\frac{W_{bag}(N_{pass} + N_{crew})}{\eta_{cargo}\rho_{cargo}g} + \frac{V_{cargo}}{\eta_{cargo}} \right) / V_{ULD} \right) \quad (6.4)$$

If the placement of outboard cargo holds is enabled, as seen in figure 4.2, cargo bays are sized much like passenger bays. If multiple cabin floors exist, the packaging algorithm attempts to fill any floors of lesser width first. The width of the resulting cargo bay is equal to that of the passenger cabin bay underneath it.

Additional cargo bays, of equal width across all floors, can be placed further outboard if additional volume is required and their width does not exceed the maximum cabin span defined. The width of each cargo bay is increased one container width at a time, until the desired volume or maximum number of containers abreast are reached.

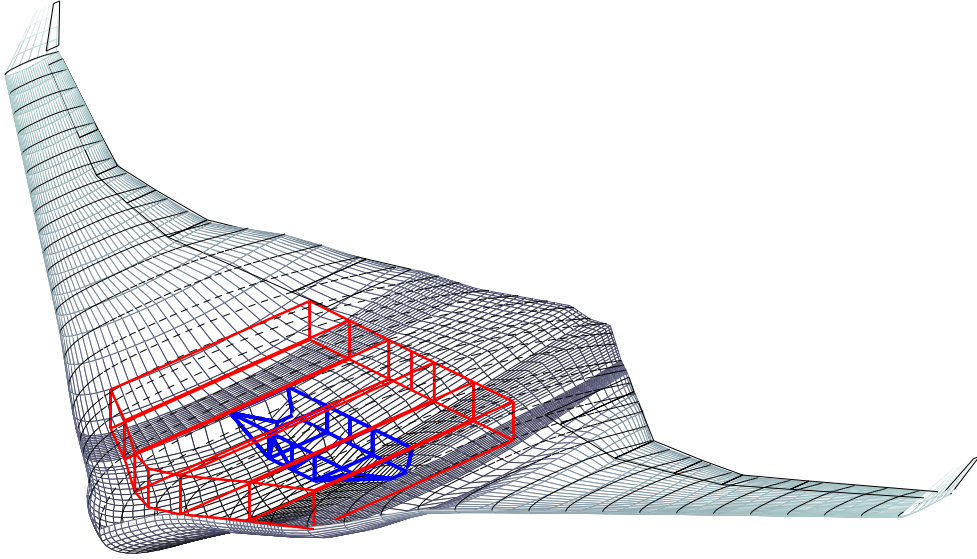


Figure 6.2: Aircraft cabin and cargo bay placements with cargo holds placed under the passenger cabin

Underfloor cargo bays, seen in figure 6.2, are sized in a different fashion. Starting from a maximum longitudinal position just forward of the step, in order to keep the centre of gravity near its intended position, the bays are extended forward until a cut-off forward position or the required volume are reached. The underbelly cargo bays span as wide as the shape of the hull allows, maintaining sufficient clearances from the bottom. The height of these bays is no longer constrained to that of the cabin and

instead equals the container height plus a two inch clearance. In cases where the ULD width varies with height, the varying width requirement is accounted for, further increasing capacity.

6.1.3 Cabin placement & centreline thickness estimation

The forward and aft limits of each cabin or cargo bay are dictated by either both structural and sectional shape constraints. As seen in figure 6.1 the cabin section is contained by the forward structural spar at the front. The rear however is constrained, not only by the initial spar location at 75% chord but by the need for a prescribed clearance (Δh) between the lower hull curve and the pressurised shell. This minimum clearance is necessary to protect the pressurised inner shell from damage and accommodate the numerous watertight compartments that must be placed above the lower surface, as dictated by seaplane survivability standards. For the upper surface it merely reflects the material thickness of structural components such as frames and the skin. It is therefore necessary that eq. (6.5) is satisfied for the entire span of the cabin and cargo sections.

$$\left(\frac{t}{c}\right)(x_{te} - x_{le}) \geq \sum_{i=1}^n h_{cabin} + \sum_{i=1}^m h_{cargo} + 2w_{cab} \left(\frac{dh}{w}\right)_{shell} + \Delta h_{u_{min}} + \Delta h_{l_{min}} \quad (6.5)$$

where n is the number of cabin floors and m is the number of under-cabin cargo floors at the spanwise location in question.

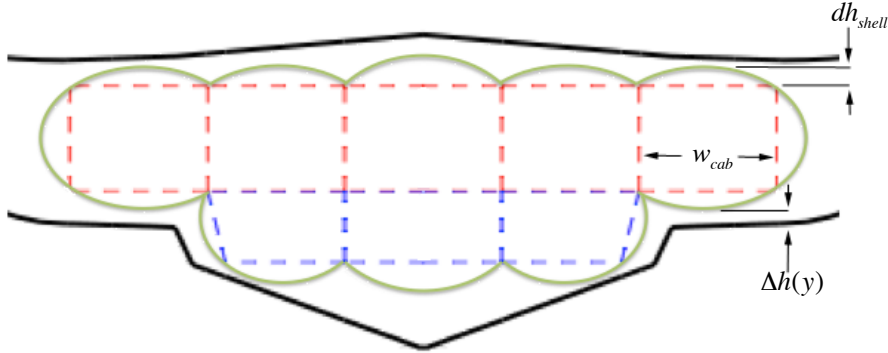


Figure 6.3: Chordwise cross-section of the aircraft at the main step, showing the cabin (red), cargo bays (blue), pressurised shell (green) and external shell (black)

Therefore for a predefined hull length (L), dictated by hydrostatic design constraints, the centreline thickness to chord ratio is:

$$\left(\frac{t}{c}\right)_o = \left(\frac{n(h_{cabin} + t_{floor})}{L(1 - \delta_u - \delta_l) + \min(dh_l(y)) - \max(dh_u(y))} \right) / (Z'_u - Z'_l) \quad (6.6)$$

where δ_u and δ_l are the upper and lower cabin position offsets, predefined as percentages of the

minimum sectional thickness, given by eqs. (6.7) to (6.9).

$$(t/c)_{\min} = \min(dh_u(y)) - \max(dh_l(y)) \quad (6.7)$$

$$dh_u(y) = Z_{\max}(y) - Z_{\max}(0) - \Delta h_{u_{\min}} - dh_{\text{shell}}(y) \quad (6.8)$$

$$dh_l(y) = Z_{\min}(y) - Z_{\min}(0) + \Delta h_{l_{\min}} + dh_{\text{shell}}(y) \quad (6.9)$$

A flowchart describing the iterative process followed in order to size and fill the aircraft's centre section can be seen in figure 6.4.

6.2 Fuel System

The fuel system must feature fuel tanks large enough to accommodate the fuel volume necessary to complete the design mission profile, as found during initial sizing. The minimum number of distinct fuel tanks is equal to the number of the aircraft's engines. As seen from the baseline configuration, this aircraft can have fuel tanks situated in three distinct areas. In the outboard wing sections, and the hull forebody and afterbody. The latter two have to abide by the same minimum bottom surface offsets that the pressured shell has to, for the same reasons.

The first step in sizing the fuel tanks is to calculate the volume of the areas chosen to house fuel. Once the maximum fuel each area can accommodate has been found, and it is found to be less than the previously allocated amount, fuel is redistributed. Given the abundance of free volume in the sea-plane's hull, compared to standard BWB designs, the forward tank is never a limiting factor, unlike the wing or aft areas.

The wing fuel tanks are integral and placed between the forward and aft spars, extending from just inboard of the wing fairing section ($y = y_{le} + 0.9dy_w\Delta Z_w$) or just outboard of the furthest outboard engine, to 95% of the available wing span. The aft fuel tank is integral to the structure, extending from the back of the cabin and the aft spar until the engine intakes. In order for space to be left for the retracted step fairing and its actuation mechanism, an appropriate minimum height constraint is also used. As the APU must also be placed aft of the cabin, the aft fuel tanks must be separated by at least the engine's diameter (d_{APU}). The forward fuel tank extends from the forward spar to the most forward under cabin component which is either the PACK unit or a belly cargo hold. The previously stated minimum clearances from the hull bottom are also maintained.

In sizing the individual fuel tanks, the aircraft outline described in chapter 5 is used. Cross sectional areas are numerically integrated until the desired fuel volume plus a predefined percentage surge excess is found. If space is still available, additional fuel tanks are created to enable further CG control and accommodate the extra amount of fuel that can be carried during ferry flights. To correct for the volume of the existing structure, 95% of the total volume is assumed to be available for fuel placed in integral tanks, while a 90% factor is used for all other fuel tank types.

6.3 Propulsion System

The aircraft designed in this synthesis uses high bypass ratio engines for propulsion. The number of engines is determined by dividing the total takeoff thrust required by the maximum allowable thrust per engine. Given the adverse effect that a low number of engines has on the takeoff performance of a seaplane, a minimum number of at least two to three engines is set.

The engines are placed along the trailing edge of the centre body at a predefined spacing (dy_{eng}) and protruding from the trailing edge by (dx_{eng}). If the resulting span of the engines is higher than the maximum allocated, they are placed closer together but spaced at least half a meter apart. If an odd number of engines is required then they are placed either side of one situated on the aircraft's centreline. Very large spacing is also avoided due to the large yawing moments that could result in case of an outboard engine failure and the BWB's relatively impaired yaw controllability.

The engine fan diameter is chosen based on the engine's design mass flow rate ($\dot{m}$) and the pre-specified fan entry Mach number (M_f). Assuming isentropic flow and using conservation of mass:

$$d_{eng} = 2 \sqrt{\frac{\dot{m}}{\pi \rho a M_f} \frac{\sqrt{2 + (\gamma - 1) M_f^2}^{\frac{\gamma+1}{\gamma-1}}}{\sqrt{2 + (\gamma - 1) M_\infty^2}^{\frac{\gamma+1}{\gamma-1}}}} \quad (6.10)$$

where a is the speed of sound and M_∞ the cruise Mach number at the engine design conditions. The length (m) and dry weight (kg) of the engine can thus be estimated using empirical sizing equations given by Raymer [67].

$$L_{eng} = 0.542 \left(\frac{d_{eng}}{0.12e^{0.04BPR}} \right)^{0.8} M_{max}^{0.2} \quad (6.11)$$

$$W_{eng} = 0.0304e^{-0.045BPR} \left(\frac{d_{eng}}{0.12e^{0.04BPR}} \right)^{2.2} \quad (6.12)$$

These equations are strictly speaking only valid for $BPR \leq 6$ but as the engines are mounted externally, the associated errors do not have a grave effect on the resulting aircraft's packaging and layout.

During flight, the flow into the fan is compressed and decelerated by the inlet. The inlet length (L_{inl}) is typically equal to the fan diameter. Assuming isentropic flow and that the change in Mach number is shared equally between the streamtube forward of the inlet and the inlet itself:

$$d_{inl} = d_{eng} \sqrt{\frac{2M_f}{\pi(M_f + M_\infty)} \left(\frac{1 + 0.05(M_f + M_\infty)^2}{1 + 0.2M_f^2} \right)^3} \quad (6.13)$$

The engines are mounted on airfoil shaped pylons. The height of placement is determined so that the engines are clear of the boundary layer that has developed over the length of the body forward of the engine's location.

Also part of the propulsion system is the Auxiliary Power Unit (APU), a jet engine used to power the aircraft's electrical systems. The APU is placed along the centreline of the aircraft, aft of the cabin. The engine is fed air by a duct extending from the aircraft's top surface to the APU inlet as does the duct for the engine's exhaust.

6.4 Fins

The baseline configuration requires a very flexible treatment of fins. All the generated aircraft feature tip-fins but they can also feature a single or pair of vertical stabilisers, aft mounted on the BWB's centre section. Each fin's geometry is defined by a set of parameters, seen in figure 6.6, and all sections are assumed to be NACA 4-series symmetrical airfoils.

As the tip fins root chord is constrained by the tip chord of the wing, it can be fully described by the aspect ratio of the fin's vertical projection ($\mathcal{R}_f$), its taper ratio (λ_f), thickness-to-chord ($(t/c)_f$), leading edge sweep (Λ_{le_f}) and angle to the vertical (γ_f). The same applies to all other fins, with the addition of S_v , the vertically projected reference surface area.

The longitudinal position of these fins is defined by the distance between the fin trailing edge at the root and the wing's leading edge. If a single central fin is required and there is an even number of engines, the angle γ_f is set to zero. Alternately the two fins are placed outboard of the furthest outboard engine, maintaining the predefined dihedral.

The resulting fin dimensions are therefore:

$$b_f = \frac{\sqrt{\mathcal{A}\mathcal{R}_f S_v}}{\cos \gamma_f} \quad (6.14)$$

$$c_{R_f} = \frac{2}{(1 + \lambda_f)} \sqrt{\frac{S_v}{\mathcal{R}_f}} \quad (6.15)$$

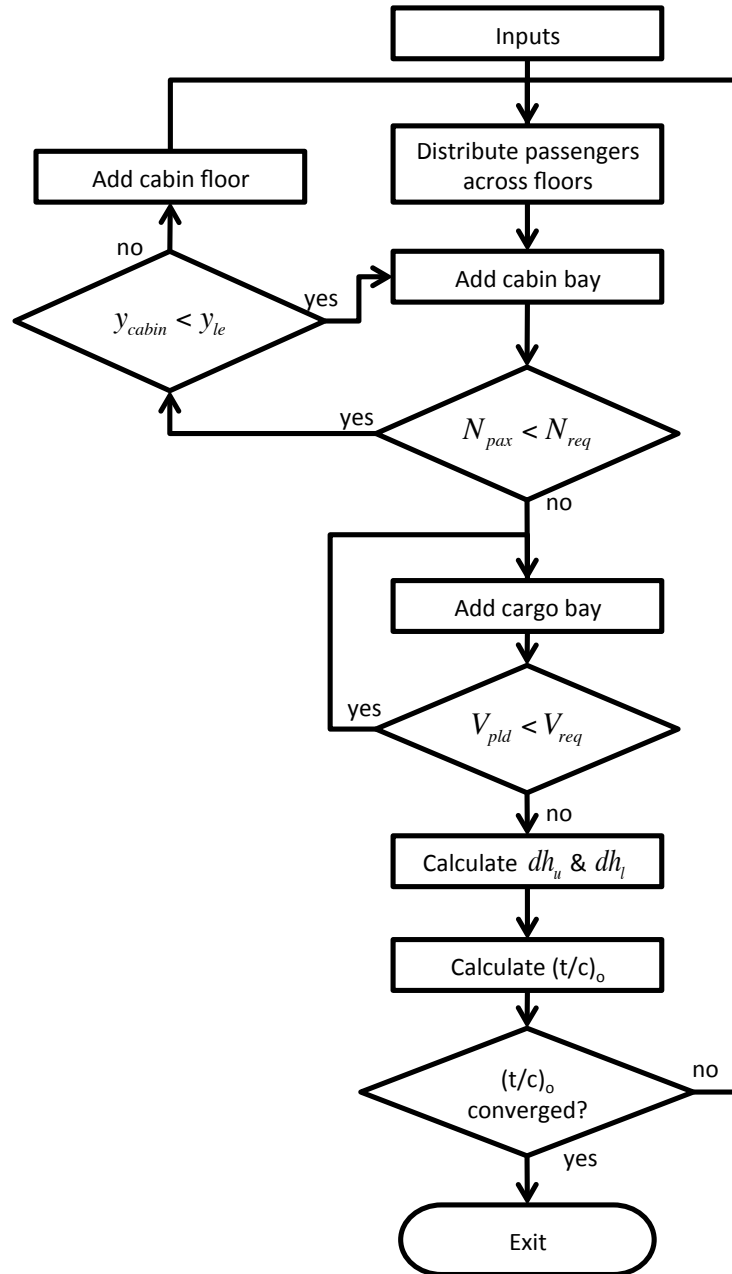


Figure 6.4: Flowchart of the aircraft's centre-section sizing process

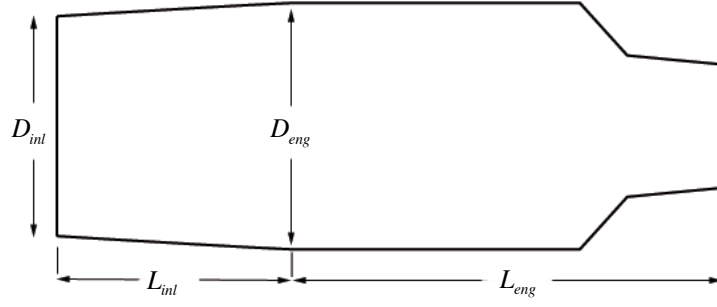


Figure 6.5: Outline of generic engine model with basic dimensions

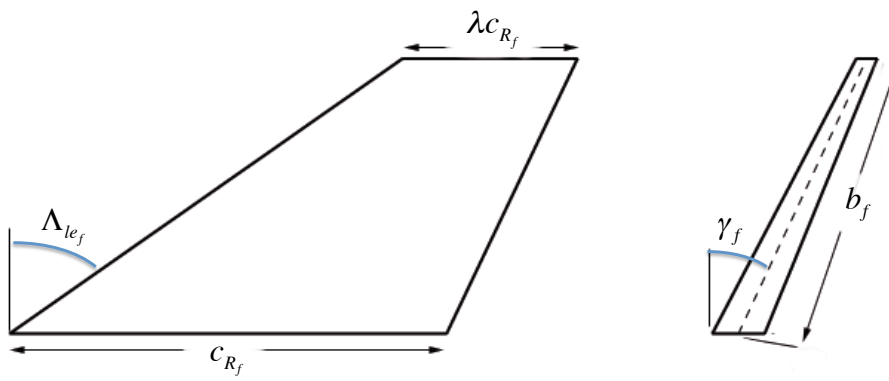


Figure 6.6: Parameters used in defining the geometry of fins

Chapter 7

Aerodynamics

The ability to correctly estimate the aerodynamic performance of an aircraft is paramount during the synthesis process. The methods described in this chapter were chosen to offer a balance between accuracy and speed, given the large number of iterations required in reaching an accurate design.

The design in question features a wing planform that can hardly be approximated by its trapezoidal equivalent and airfoil sections that vary substantially with the BWB's span. Consequently the use of some well established empirical methodologies and formulae was deemed unsuitable. For the estimation of aerodynamic pressure loads, the use of a Vortex Lattice (VLM) solver was opted for instead.

Also presented herein are the methods used to optimise the wing lift distribution, estimate subsonic drag, the transonic drag rise and the effect of control surfaces. The effects of introducing a step along the lower surface of an airfoil section are discussed and a novel method for calculating them is presented.

7.1 Pressure loads

The design of a BWB requires not only the accurate determination of the wing's lifting characteristics but also the position of the centre of pressure. Conventional tube-and-wing aircraft feature a high aspect ratio wing whose lift can be easily estimated using eq. (7.1) based on the sectional lift characteristics.

$$C_{L\alpha} = \frac{2\pi\mathcal{R}}{2 + \sqrt{4 + \frac{4\pi^2\mathcal{R}^2}{C_{l_\alpha}^2} \left(1 + \frac{\tan^2 \Lambda_{max}}{1 - M^2}\right)}} \quad (7.1)$$

The aircraft's centre of pressure is then determined based on assumptions about the centre of pressure location on each trapezoidal surface and their relative position along the aircraft's length. Applying this method to a flying wing or BWB is not feasible therefore a computational approach to solving

the problem is necessary.

The sheer level of computational cost made the use of CFD or three-dimensional Euler solvers impossible at this early stage in design. Potential flow solutions were consequently deemed the most appropriate. A three-dimensional panel code can account not only for planform but also thickness and camber variations, however a large number of panels must be used to accurately represent the lifting surfaces. The least computationally taxing approach is the use of a lifting line formulation, however the effects of sectional camber and sweep would not be accounted for. The vortex lattice formulation was thus identified as offering the most favourable mixture of accuracy versus computational cost.

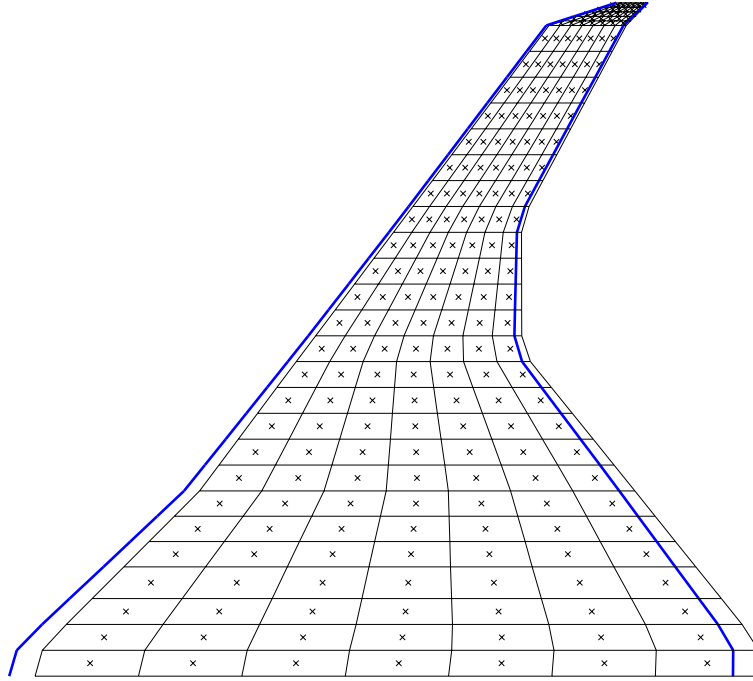


Figure 7.1: Top view of lifting surface showing the placement of vortex rings (black) and collocation points (x) relative to the wing's leading and trailing edges (blue)

The vortex lattice method, described in more detail by Katz and Plotkin [43], approximates bodies as lifting surfaces of zero thickness following the wing's camber line, discretised using a mesh of $n \times m$ panels. The flow about the lifting surface is modelled by placing vortex ring elements of equal size to the original panel, 25% of the panel length downstream. A vortex ring element is a quadrilateral made of four finite linear vortex filaments of equal circulation Γ , placed end to end. Collocation points are then placed at the centre of each vortex ring or at 75% of each panel's chord. The geometry generated by this placement of vortex ring elements and collocation points relative to the wing leading

and trailing edges can be seen in figures 7.1 and 7.2.

To satisfy the Kutta condition, which requires that circulation is zero along the wing's trailing edge, an additional vortex ring must be attached to the trailing edge of the aft most vortex ring. This wake panel extends to infinity parallel to the local free stream velocity and has a vortex strength equal to that of the panel it is attached to. This alignment of the wake vortex offers a more accurate representation of the physical flow than in traditional VLM implementations, where it follows the wing angle at the trailing edge.

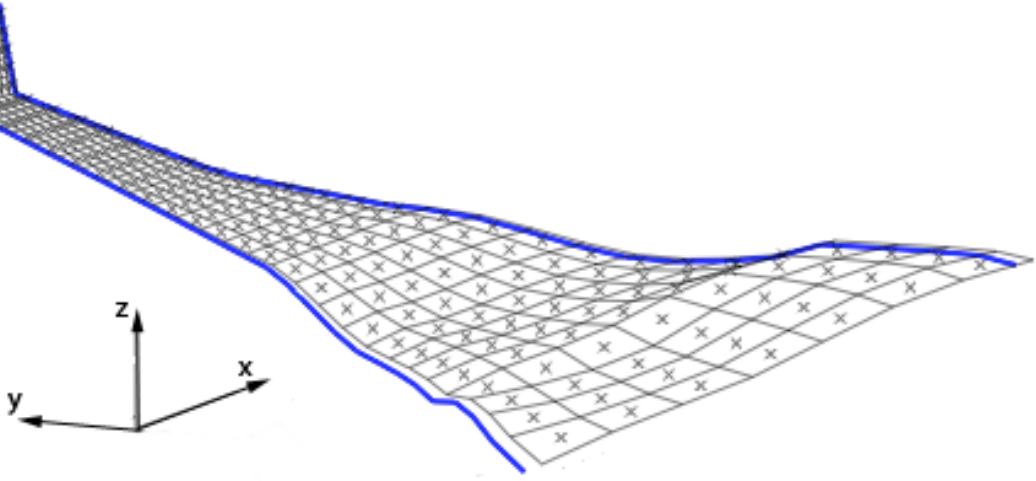


Figure 7.2: Forward side view of lifting surface showing the way varying camber and dihedral are represented using a mesh of 30×7 panels

The strength of each vortex ring is found by imposing a Neumann boundary condition at each collocation point, requiring that the net velocity normal to the surface is zero. Therefore the Neumann condition for the j^{th} collocation point is.

$$\sum_{i=1}^{n(m+1)} (\mathbf{u}_i \cdot \hat{\mathbf{n}}_j) = -\mathbf{U}_j \cdot \hat{\mathbf{n}}_j \quad (7.2)$$

where $\hat{\mathbf{n}}_j$ is the unit vector normal to each panel. The local velocity ($\mathbf{U}_j$) at the collocation point, with coordinates $\mathbf{c} = [x_c, y_c, z_c]$ and with rates of rotation $[p, q, r]$ with respect to the stability axes and about the reference point $[x_{ref}, y_{ref}, z_{ref}]$ is:

$$\mathbf{U}_j = \mathbf{U}_\infty + \begin{bmatrix} p \\ -q \\ r \end{bmatrix} \times \begin{bmatrix} x_c - x_{ref} \\ y_c - y_{ref} \\ z_c - z_{ref} \end{bmatrix} \quad (7.3)$$

The velocity induced by the i^{th} vortex ring ($\mathbf{u}_i$) can be found as the sum of the velocities induced by

each vortex filament, spanning from $\mathbf{p}_k$ to $\mathbf{p}_{k+1}$, as given by the Biot-Savart Law.

$$\mathbf{u}_i = \sum_{k=1}^4 \frac{\Gamma}{4\pi|\mathbf{r}_1 \times \mathbf{r}_2|^2} \left(\frac{\mathbf{r}_o \cdot \mathbf{r}_1}{|\mathbf{r}_1|} - \frac{\mathbf{r}_o \cdot \mathbf{r}_2}{|\mathbf{r}_2|} \right) \quad (7.4)$$

where

$$\mathbf{r}_o = \mathbf{p}_{k+1} - \mathbf{p}_k$$

$$\mathbf{r}_1 = \mathbf{c} - \mathbf{p}_k$$

$$\mathbf{r}_2 = \mathbf{c} - \mathbf{p}_{k+1}$$

Solving the system of simultaneous linear equations given by eq. (7.2), the aircraft's vorticity distribution is found. The effects of compressibility are modelled by applying the Prandtl-Glauert correction to the vortex ring strengths found, as suggested by Melin et al. [54].

$$\Gamma_c = \frac{\Gamma_i}{\sqrt{1 - M^2}} \quad (7.5)$$

Having found the vortex strengths necessary to satisfy the boundary conditions set, the force ($\mathbf{F}$) and resulting moments generated by each vortex vortex ring is given by using the Kutta Joukowski theorem (7.6) to each individual vortex filament.

$$\mathbf{F}_j = -\rho \Gamma_j \sum_{k=1}^4 [(\mathbf{p}_{k+1} - \mathbf{p}_k) \times \mathbf{U}_{tot}] \quad (7.6)$$

$$\mathbf{U}_{tot} = \mathbf{U}_{\infty} + \begin{bmatrix} p \\ -q \\ r \end{bmatrix} \times \left(\frac{\mathbf{p}_{k+1} + \mathbf{p}_k}{2} - \begin{bmatrix} x_{ref} \\ y_{ref} \\ z_{ref} \end{bmatrix} \right) + \sum_{i=1}^{n(m+1)} \mathbf{u}_i \quad (7.7)$$

If no sideslip and zero rates of roll and yaw are defined, the spanwise force distribution can be assumed symmetrical about the centreline. In such a case the computational cost can be minimised by simply mirroring the wing about the x-z plane rather than using double the number of panels. The use of a VLM code is also highly advantageous as the effects of ground proximity during takeoff and landing, known as ground effect, can be easily and accurately modelled by simply mirroring the aircraft about the ground plane. In both cases, the effect of the mirrored vortex rings must also be considered when evaluating eq. (7.2).

The VLM implementation in this synthesis is used to analyse the entire aircraft as a whole, excluding however the engines and their supporting pylons. This is done so that the effect each lifting surface's flow field has on another's can be accounted for.

Assuming a linear variation of both lift and pitching moment with angle of attack (α), the lift (C_L) and moment (C_M) coefficients of the entire aircraft about x_{ref} and at a flight mach number (M) due to the pressure loads can be interpolated as:

$$C_L = \left(C_{L_o} + \frac{dC_L}{d\alpha}(\alpha - \alpha_o) \right) \sqrt{\frac{1 - M_o^2}{1 - M^2}} \quad (7.8)$$

$$C_M = \left[C_{M_o} + \frac{dC_M}{d\alpha}(\alpha - \alpha_o) + \frac{x_{ref} - x_o}{\bar{c}} \left(C_{L_o} + \frac{dC_L}{d\alpha}(\alpha - \alpha_o) \right) \right] \sqrt{\frac{1 - M_o^2}{1 - M^2}} \quad (7.9)$$

where C_{L_o} and C_{M_o} , estimated with respect to the point x_o , are the coefficients given by the VLM at an angle of attack α_o and mach number M_o . The derivatives $dC_L/d\alpha$ and $dC_M/d\alpha$ are estimated by reevaluating the aircraft's aerodynamic characteristics at $\alpha_o + d\alpha$ and using a forward finite difference method. The lift induced drag component, as predicted by the VLM can also be interpolated using:

$$C_{D_i} = KC_L^2 = (C_{D_i})_o \left(\frac{C_L}{C_{L_o}} \right)^2 \quad (7.10)$$

The methods described above have been shown to produce accurate results for Mach numbers below 0.7 and angles of attack in the linear part of the lift curve. A fundamental assumption is that the wing is thin, something that does not hold for the vast majority of wing centre sections necessary to fulfil both shape and packaging constraints.

7.2 Lift Distribution & Optimum Twist

Finite wings experience lift induced drag due to the effect that vortices, shed from the wing tips, have on the flow field around the wing itself. As shown by eq. (7.10) the induced drag's magnitude is proportional to the square of the lift coefficient by factor K , defined as:

$$K = \frac{1}{\pi e \mathcal{R}} \quad (7.11)$$

, where the Oswald efficiency (e) is a measure of the induced drag generated by the desired planform, relative to that of the optimum section of equal aspect ratio. The efficiency factor is maximised when a wing's spanwise lift distribution is elliptic.

For a wing of any planform, an elliptic lift distribution can be approximated by twisting the wing. As the lift distribution varies with lift coefficient, the twist is optimised for mid cruise. In the case of a

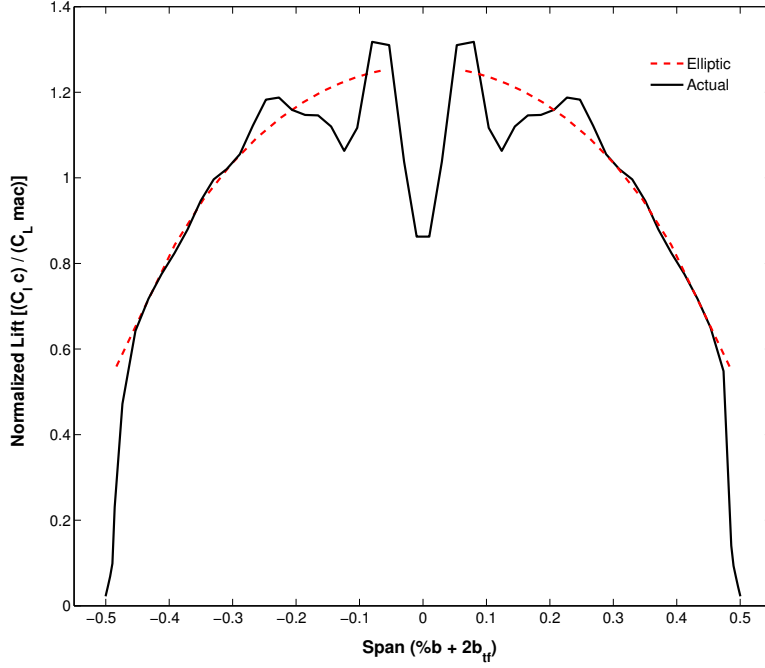


Figure 7.3: Lift distribution achieved with the use of twist compared to the desired elliptical distribution

seaplane BWB however certain limitations are imposed on the span that can be twisted and the twist distribution itself. In order to preserve the shape of the hull, the wing can only be twisted outboard of the hull rise. As seen in figure 7.3, the hull section usually produces lower levels of lift than a typical airfoil shaped section would.

The peaky lift distribution produced without the use of twist was found to require a very irregular use of twist, going from -6 to 6 degrees over a very short span, especially in sections that must house cabin and cargo bays. To avoid this, the twist distribution was constrained with a low pass filter to smoothly vary with span.

$$l_i = \frac{2W}{\pi(b/2 + b_{tf})} \sqrt{1 - \left(\frac{y}{b/2 + b_{tf}} \right)^2} \quad (7.12)$$

The twist itself is found by trying to iteratively match an elliptical lift distribution (7.12) that produces the desired amount of lift at the design point. A number of control points (n_{tw}), 15 in this implementation, are used. Given that the lift variation is most pronounced for the inboard sections, where the airfoil sections vary most, the control points are spaced using a cosine distribution.

$$(y_c)_i = B \left(\frac{1}{2} + \frac{\Delta Z_{hr}}{\tan \gamma_{hr}} \right) + \left[\frac{b}{2} - B \left(\frac{1}{2} + \frac{\Delta Z_{hr}}{\tan \gamma_{hr}} \right) \right] \left(1 - \cos \left(\frac{\pi(i-1)}{2(n_{tw}-1)} \right) \right) \quad (7.13)$$

To speed up convergence a sectional lift curve slope of 2π was assumed, enabling the necessary twist

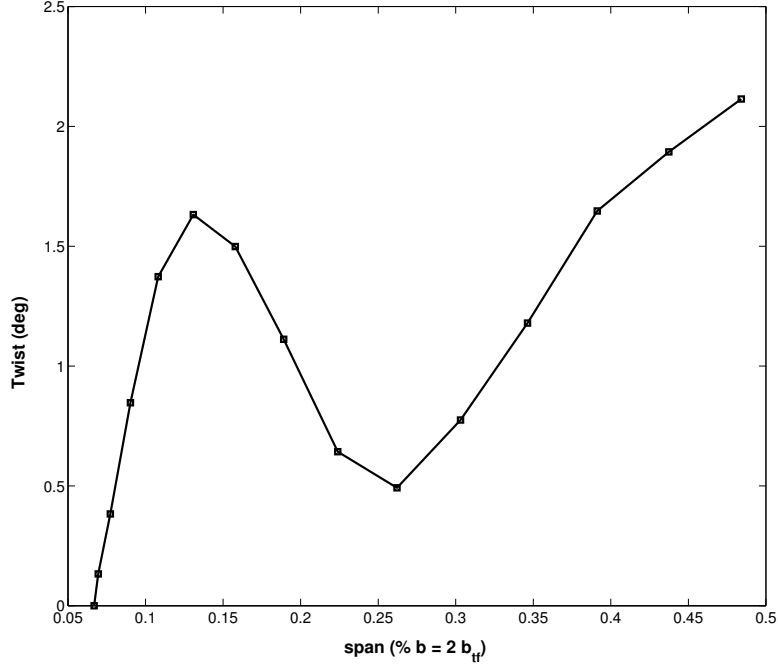


Figure 7.4: Twist variation necessary to achieve the lift distribution shown in figure 7.3, also showing control point distribution

increment to be determined accordingly. A representative lift distribution and the twist distribution necessary to achieve it can be seen in figures 7.3 and 7.4.

7.3 Viscous Effects

The aerodynamic analysis methods described in section 7.1 assumes inviscid flow, therefore the viscous contributions to drag must be accounted for otherwise. The component buildup method, presented by Raymer [67], has been found to produce rather accurate results, by estimating the aerodynamic frictional loads experienced by each aircraft component and summing them up.

$$C_{D_o} = \frac{\sum(\bar{C}_f S_{wet} F F_c Q_c)}{S_{ref}} \quad (7.14)$$

The mean friction coefficient ($\bar{C}_f$) is found as the weighted average of the laminar and turbulent boundary layer values.

$$\bar{C}_f = p_l C_{f_{lam}} + (1 - p_l) C_{f_{turb}} \quad (7.15)$$

where the flat plate friction coefficient for a laminar boundary layer is given by the Blasius equation:

$$C_{f_{lam}} = \frac{1.328}{\sqrt{Re_c}} \quad (7.16)$$

For a turbulent boundary layer in subsonic flow, Raymer [67] suggests the use of:

$$C_{f_{turb}} = \frac{0.455}{(\log_{10} Re_c)^{2.58} (1 + 0.144 M^2)^{0.65}} \quad (7.17)$$

The average is weighed by the percentage of the wetted surface subject to a laminar boundary layer (p_l) roughly found by:

$$p_l = \min(1, Re_{trans}/Re_c) \quad (7.18)$$

where the assumed transition Reynolds number $Re_{trans} = 5 \times 10^5$ and Re_c is the sectional chord based Reynolds number.

The shape of each individual component is accounted for using an empirical form factor correction.

For wings, fins and pylons:

$$FF_c = \left[1 + \frac{0.6}{(x/c)_m} \left(\frac{t}{c} \right) + 100 \left(\frac{t}{c} \right)^2 \right] [1.34 M^{0.18} (\cos \Lambda_m)^{0.28}] \quad (7.19)$$

where $(x/c)_m$ is the sectional maximum thickness point and Λ_m is the maximum thickness sweep angle. For the nacelles and engines:

$$FF_c = 1 + \frac{0.35 D_{eng}}{L_{inl} + L_{eng}} \quad (7.20)$$

For the wing section, eq. (7.14) is evaluated numerically for thin successive strips along the wing's span. To account for the aerodynamic effect of the hull's shaping, an interference factor (Q_c) of 1.1 is used for $y \leq B(0.5 + \Delta Z_{hr}/\tan \gamma_{hr})$ and unity thereafter. For fins not acting as extensions to the wing interference with the fuselage is assumed to lead to a 4% increase, for pylons approximately 10% and for body mounted engines:

$$Q_c = \max(1, 1.5 - 0.2h/D_{eng}) \quad (7.21)$$

where h is the pylon span.

7.4 Transonic Effects

Modern airliners fly in the transonic regime to take advantage of the associated increase in lift to drag ratio. Flying at transonic velocities however is associated with sharp rise in drag, The onset of wave drag starts at the drag divergence Mach number (M_{dd}), which is determined by both the wing's planform and cross sectional shape. In the preliminary design stage, the sectional M_{dd} can be found using the Korn equation [34].

$$M_{dd} = \frac{\kappa_A}{\cos \Lambda_{0.5}} - \frac{t/c}{\cos^2 \Lambda_{c/2}} - \frac{C_l}{10 \cos^3 \Lambda_{0.5}} \quad (7.22)$$

where κ_A is an empirical technology factor, equal to 0.86 for conventional airfoils and 0.95 for supercritical ones. The technology factor is chosen based on the performance a particular airfoil section could have if optimised with minor alterations, rather than the performance of the particular geometry used in the synthesis. The sectional lift coefficient (C_l) is obtained from the wing's lift distribution and $\Lambda_{c/2}$ is the section's local semi-chord sweep. The free stream Mach number at which the flow over the wing's cross section reaches sonic speeds is then empirically given by:

$$M_{crit} = M_{dd} - \sqrt[3]{\frac{0.1}{80}} \quad (7.23)$$

The overall aircraft's drag divergence Mach number is then taken to be equal to the minimum sectional value found.

The variation of wave drag (C_{D_w}) with Mach number can then be estimated. For this synthesis two distinct methods have been implemented for the user to choose. The method presented by Raymer [67], empirically constructs a drag vs. Mach number curve based on the volumetric wave drag ($C_{D_{w-v}}$), which can be approximated using eqs. (7.24) and (7.25).

$$C_{D_{w-v}} = \frac{1}{2\pi S_{ref}} \int_0^L \int_0^L A''(x_1)A''(x_2) \ln \left(\frac{1}{|x_1 - x_2|} \right) dx_1 dx_2 \quad (7.24)$$

$$C_{D_w} = \begin{cases} 0 & \text{if } M \leq M_{crit} \\ 0.002 \tan(3.125\pi(M - M_{crit})) & \text{if } M_{crit} < M \leq M_{crit} + 0.08 \\ 0.002 + (0.5C_{D_{w-v}} - 0.002) \left[\tan \left(\frac{\pi}{4} \frac{M - M_{crit} - 0.08}{0.02 - M_{crit}} \right) \right]^{1.3} & \text{if } M_{crit} + 0.08 < M \leq 1 \end{cases} \quad (7.25)$$

The volume based wave drag is found based on the lengthwise variation of spanwise cross sectional area ($A(x)$) and is greatly increased if sharp changes or discontinuities are present. The nature of eq. (7.24) however makes its numerical evaluation extremely error prone, it is therefore best evaluated using the numerical methods presented by Sheppard [82].

Gur et al. [34] has shown that for $M < 0.9$ accurate results can also be obtained using a finite strip method, where the sectional drag rise of multiple spanwise strips, each of planrform area S_i , is summed up by:

$$C_{D_w} = \sum \frac{S_i}{S_{ref}} C_{d_w} \quad (7.26)$$

and the sectional drag rise is empirically estimated as:

$$C_{d_w} = \begin{cases} 0 & \text{if } M \leq M_{crit} \\ 20(M - M_{crit})^4 & \text{if } M > M_{crit} \end{cases} \quad (7.27)$$

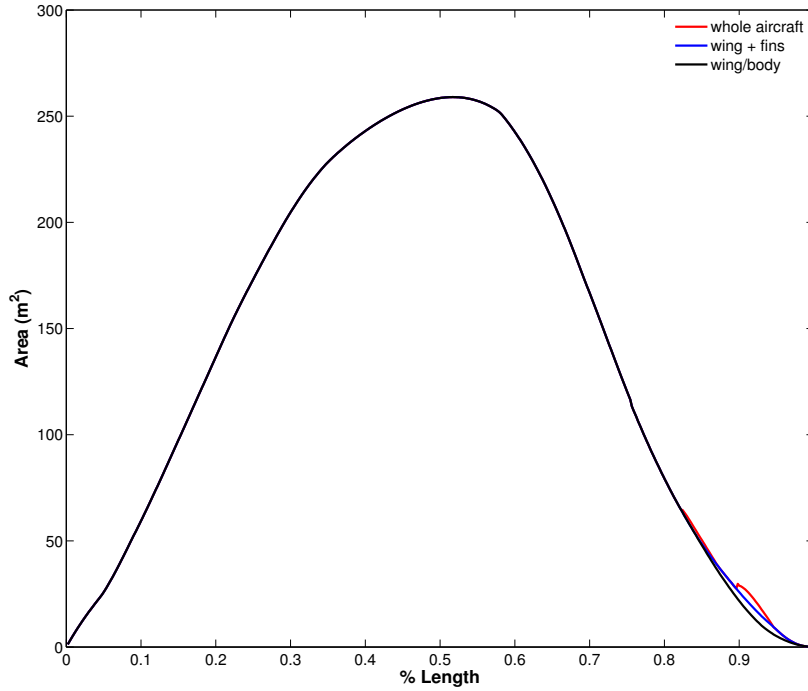


Figure 7.5: Aircraft lengthwise cross-sectional area distribution with a faired step

Use of this method has the added advantage of being somewhat applicable when the effect of yaw and roll rates is to be investigated.

7.5 Control Surfaces & High Lift Devices

The effect of actuating control surfaces and high lift devices on the lift (ΔC_L), moment (ΔC_M) and drag (ΔC_D) characteristics of the aircraft is determined using the empirical methods in DATCOM, presented in graphical format by Roskam [73]. For use within the synthesis algorithm, the graphs were digitised and the necessary value were found using multi-linear interpolation.

7.6 Maximum Lift

The stall characteristics of the aircraft, including meeting the approach speed requirement discussed in section 4.2.3.1, depend on the maximum lift coefficient ($C_{L_{max}}$) achievable by the aircraft. The aircraft is considered to have reached its maximum lift capability as soon as an individual cross section reaches it's maximum lift ($C_{l_{max}}$). The wing lift distribution and sectional maximum lift coefficients at stall can be seen in figure 7.6.

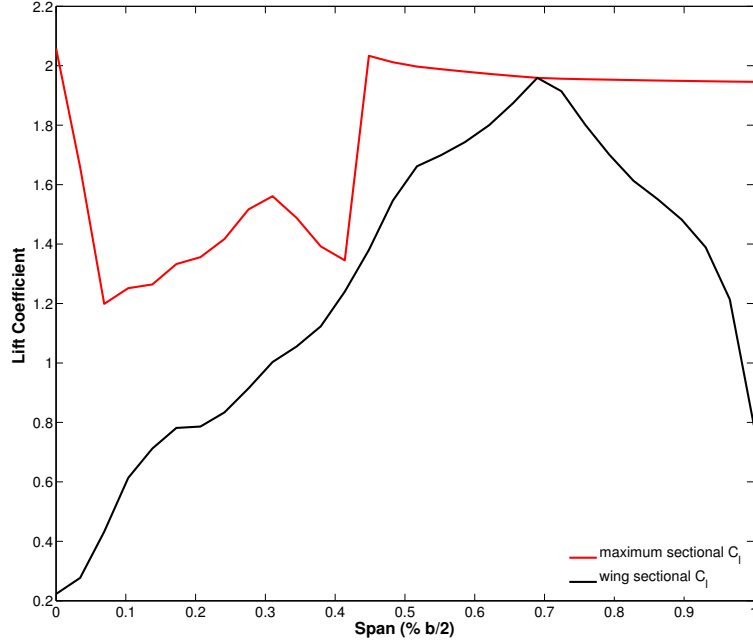


Figure 7.6: Lift distribution & sectional $C_{l_{max}}$ variation at the stall angle of attack

For known airfoils sections, the sectional maximum lift coefficient can be predefined for $Re = 9 \times 10^6$ based on experimental data. For $y < y_{le} + \Delta y_w \Delta Z_w$ however the airfoil shapes used cannot be known beforehand and thus their $C_{l_{max}}$ must be estimated. The use of a panel code using viscous-inviscid matching to predict boundary layer separation (such as Xfoil or ESDU's VGK) was considered, however the resulting values were rather inaccurate, especially at high sectional Reynolds numbers. The use of the empirical methods used in DATCOM was opted for instead.

The DATCOM method relates the sectional maximum lift coefficient at $Re = 9 \times 10^6$ to the magnitude and chord wise position of maximum camber, the thickness to chord ratio and mach number. All the above have been found to have varying effects depending on the airfoil's leading edge shape, described by the parameter Δy which equals the difference in airfoil height to chord ratio between 6% and 0.15% of the local chord length. The values obtained, both estimated and predefined, can be subsequently corrected to the correct sectional Reynolds number. The base value and corrections are again found from the digitised graphs using multilinear interpolations, except for the Re correction where a logarithmic relation was deemed more appropriate.

Also obtained using the DATCOM method are the maximum wing lift coefficient increments due

to high lift devices. Once determined using the empirical equations and associated graphs, these values are simply added to the value of $C_{L_{max}}$ found for the clean aircraft.

7.7 Step Effects

Early on in the conceptual design process, the aerodynamic penalties of using a stepped hull were identified and the use of a retractable fairing in flight was decided. A method for determining the effects of the step is however still necessary when estimating takeoff length and to ensure that rate of climb requirements during takeoff and landing are met.

As discussed in section 3.3 the methods currently suggested for estimating the drag penalty of a normal step fall short of the required level of accuracy. As steps affect the pressure distribution of an airfoil, the sectional lift and moment coefficients are also affected. The idea that a stepped airfoil can actually improve the flying characteristics of an aircraft was proposed by Kline [45] and the Kline-Fogleman airfoil has shown promise in radio controlled model aircraft applications. Most recently Finaish and Witherspoon [31] investigated the effect of steps situated either on the top or bottom surface of a NACA 0012 airfoil. Their computational results showed that in most cases a step on the airfoil's bottom surface resulted in a reduction of sectional L/D, the penalty becoming more severe the higher the step height was. The location of the step was also found to affect the section's performance characteristics.

Building an empirical model to predict the effect that a single stepped hull has on the blended airfoil sections of the centre body, should ideally rely on experimental data. Unfortunately existing experiments on rear facing backstops always deal with geometries featuring surfaces normal to each other and do not consider the effects of pressure gradients, as they would exist on the surface of an airfoil. Computational Fluid Dynamic (CFD) results were deemed a reasonable substitute however, allowing the accurate aerodynamic analysis of multiple airfoils. A total of 32 different airfoil shapes were thus analysed at 5 distinct angles of attack using the commercial software package FLUENT. The airfoils were produced using the method described in section 5.4.1, each airfoil's parameters chosen at random from the limits given in table 7.1. The leading edge radius used is related to the chosen thickness to chord ratio based on the relation used by NASA for the SC(2) series of supercritical airfoils.

$$R_u = R_l = 1.477 \left(\frac{t}{c} \right)^2 + 0.005 \quad (7.28)$$

	Range		Units
t/c	0.1	0.2	
h_s/c	0.005	0.030	
L_f/L	0.4	0.6	
L_{flat}/L	0.06	0.20	
X_{co}	0.75	1.00	
δ_k	4.0	9.0	deg.
τ_o	0.0	1.5	deg.
Z'_u/Z'_l	0.65	1.50	
X_u	0.35	0.50	
Z_{XX_u}	-0.1	0.0	
Z'_{te}	-0.30	0.30	
ΔZ_{te}	0.001	0.009	
α_{te}	-15.0	15.0	deg.
β_{te}	20.0	35.0	deg.
Re	7×10^6	9×10^8	
M_∞	0.2	0.7	

Table 7.1: Range of parameters used to design test stepped airfoils

Each airfoil was tested at angles of attack of -5, -2, 1, 3 and 6 degrees at a randomly chosen Mach and Reynolds number. CFD analysis works by solving the Reynolds Averaged Navier-Stokes (RANS) and energy equations using the finite volume method. Given the range of Reynolds numbers used, boundary layers are assumed to be fully turbulent and the $k-\omega$ SST model was chosen, based on previous knowledge of its applicability to airfoil problems and the findings of Kim et al. [44] on the relative accuracy of turbulence models in modelling the flow behind a backstop. Based on past experience, a turbulence intensity of 3% and viscosity ratio of 5 were chosen. A structured C-grid mesh (fig. 7.7) with over 1.9×10^5 cells was used as the results it produced for a NACA 0012 airfoil at the desired angles resulted in less than 2% errors in L/D with respect to the experimental data quoted by Abbott and VonDoenhoff [2]. To ensure that the flow around the flow field is not affected by the control volume boundaries, a control volume 24 chord lengths wide and $32c$ long was chosen,

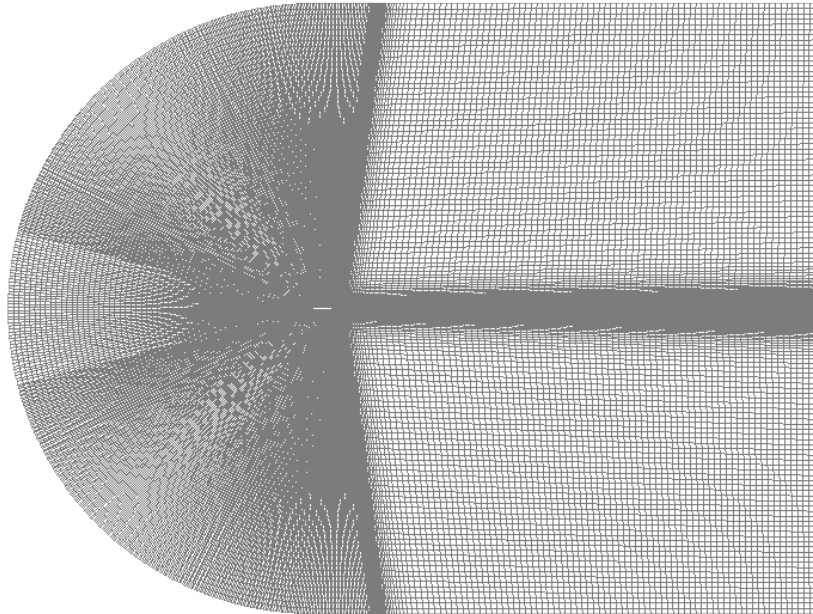


Figure 7.7: View of the C-grid used for the analysis of the family of stepped airfoils

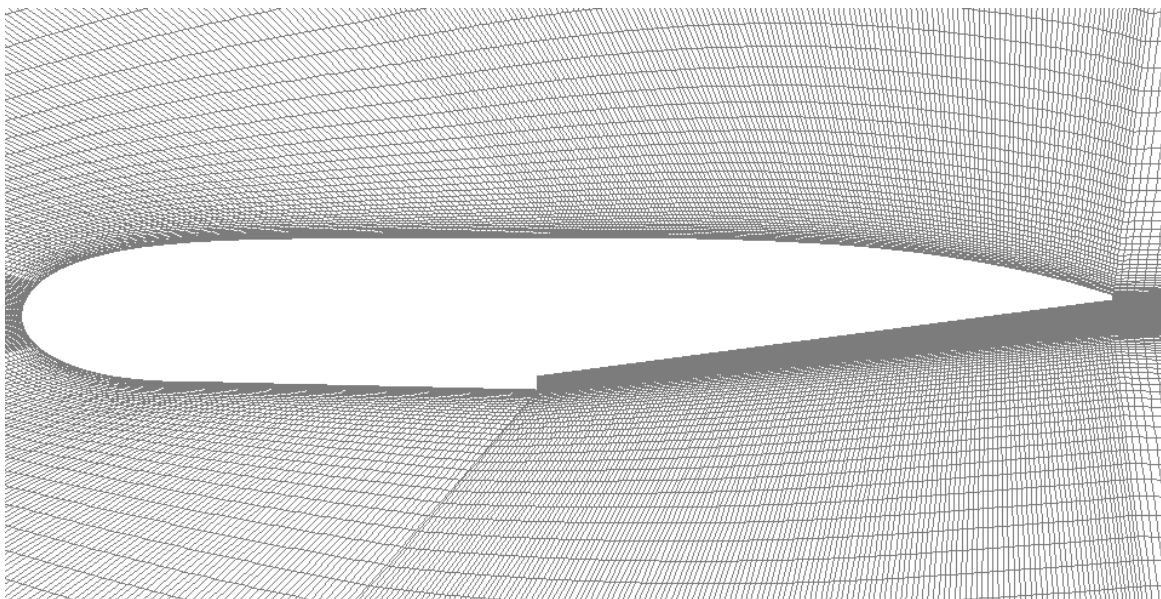


Figure 7.8: Detail of the meshing structure near the airfoil surface

with the airfoil placed roughly at the centre.

As seen in figure 7.8, the airfoil itself is meshed with a minimum of 225 cells on the top surface, 125 on the bottom surface forward of the step and an additional 150 aft. Based on previous experience, a dense grid was used to accurately predict the flow in the recirculation region aft of the step, leading to the use of over 6×10^3 cells. The cell heights on the airfoil surface were adjusted throughout the analysis in order for $30 < y^+ < 300$, as required by the turbulence model. The adjustments were done within the solver by repeatedly quartering the cells nearer the airfoil surface until the constraint was met.

An average of about 10000 iterations were required for the predicted values for C_l and C_d to converge. Convergence was however not possible in a limited number of high mach number cases at larger angles of attack and the results were thus omitted.

Figure 7.10 shows the x-velocity distribution about test airfoil 26 at an angle of attack of 3 degrees. It should be noted that in all cases the angle of attack is referenced to the airfoil baseline ($z = 0$) rather than the chord. On the top surface the acceleration of the airflow to sonic velocities and subsequent shock can be seen to increase the local boundary layer thickness. On the lower surface, the flow becomes separated aft of the step. The recirculation region and the eventual reattachment of the flow can be seen from the airfoil's streamlines (fig. 7.9).

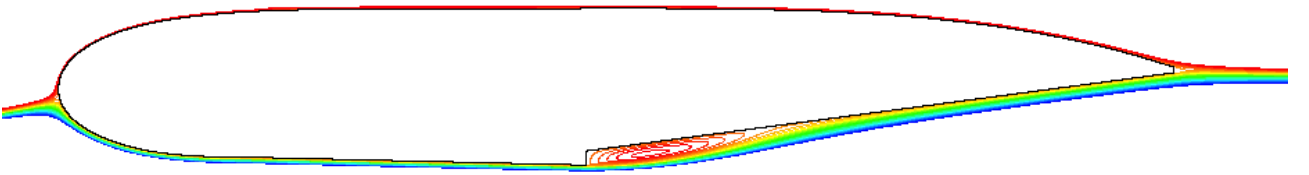


Figure 7.9: Streamlines of the flow around test airfoil 26, $\alpha = 3$, $M_\infty = 0.643$, $Re = 8.811 \times 10^8$

The results of all the numerical computations performed can be seen in tables C.3 to C.8. Comparing the CFD results to the lift, drag and moment coefficient of the same airfoil with a 9:1 step fairing, the incremental effect of the step can be determined. The aerodynamic properties of the airfoil with a faired step were found using potential flow solvers with viscous corrections both compressible and incompressible. The panel code Xfoil chosen as it could be easily incorporated into the synthesis process.

The design parameters that most affected the aerodynamic properties of a stepped airfoil were

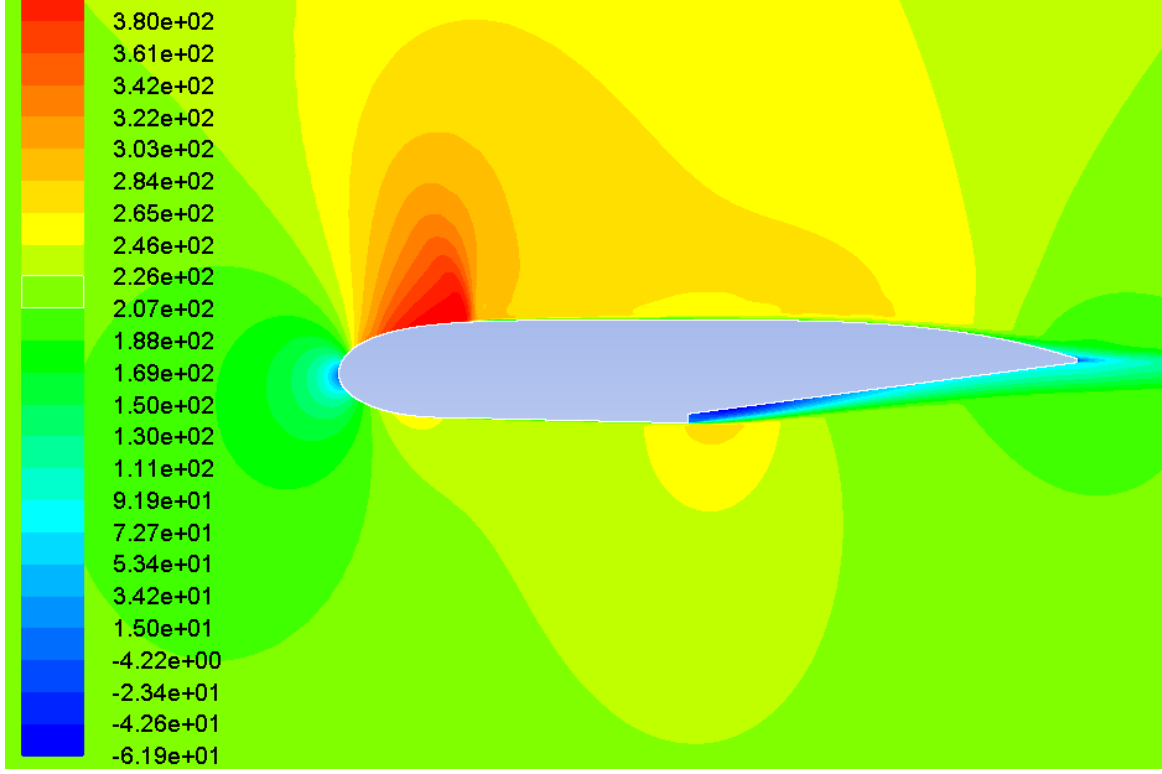


Figure 7.10: Variation of x-velocity component (m/s) around test airfoil 26, $\alpha = 3$, $M_\infty = 0.643$, $Re = 8.811 \times 10^8$

determined, by performing a sensitivity analysis, to be the step height, afterbody keel angle and thickness to chord ratio. The chord wise position of the step, for the range investigated, was not found to have a substantial effect. The Mach number and sectional Reynolds number were also found to have a substantial effect. Using multivariate linear regression for each set of angles of attack, the drag increment due to a step is given by eq. (7.29) and the coefficient necessary can be seen in table 7.4.

$$\begin{aligned} \frac{\Delta C_d}{h/c} = & a_0 + a_1 \tan(\delta_k - \tau_o) + \frac{a_2}{Re^{0.2}} + a_3 M + a_4 M \tan(\delta_k - \tau_o) + \\ & + a_5 \frac{h}{c} \tan(\delta_k - \tau_o) + a_6 M^3 + a_7 \frac{h}{c} + a_8 M \frac{h}{c} + a_9 \frac{t}{c} + a_{10} \frac{h}{c} \frac{t}{c} \end{aligned} \quad (7.29)$$

The change in lift coefficient was found to be best represented in terms of the change in sectional L/D. Unlike the drag equation, the lift to drag and moment equations must also account for the performance of the faired airfoils, as shown by eqs. (7.30) and (7.31).

$$\begin{aligned} \frac{\Delta(l/d)}{h/c} = & a_0 + a_1 \frac{l}{d} + a_2 \tan(\delta_k - \tau_o) + a_3 M + a_4 \log_{10} Re + a_5 M \tan(\delta_k - \tau_o) + \\ & + a_6 \frac{h}{c} + a_7 M \frac{h}{c} + a_8 \left(\frac{h}{c}\right)^2 + a_9 \frac{t}{c} + a_{10} \frac{t}{c} \frac{h}{c} \end{aligned} \quad (7.30)$$

$$\begin{aligned} \frac{\Delta C_m}{h/c} = & a_0 + a_1 \frac{t}{c} + a_2 C_m + \left(a_3 + a_4 \frac{h}{c} + a_5 M\right) \tan(\delta_k - \tau_o) + a_6 \log_{10} Re + \\ & + a_7 M^3 + a_8 M + a_9 M \frac{h}{c} + a_{10} \tan \tau_o \end{aligned} \quad (7.31)$$

	$\alpha = -5 \text{ deg.}$	$\alpha = -2 \text{ deg.}$	$\alpha = 1 \text{ deg.}$	$\alpha = 3 \text{ deg.}$	$\alpha = 6 \text{ deg.}$
R^2	0.820380	0.888914	0.931289	0.841516	0.879704
a_0	-2.747844	-0.561049	-0.148560	0.634217	2.321323
a_1	2.376125	0.888219	1.700132	2.891751	5.813859
a_2	22.127428	5.758600	5.378261	3.311629	-2.166009
a_3	0.800501	0.103863	-0.142001	-1.812508	-7.246545
a_4	-3.274116	2.063672	2.671675	0.983778	8.418057
a_5	214.669183	139.945444	31.271092	-39.378338	-377.166306
a_6	10.645608	3.303789	1.754738	4.804811	17.234663
a_7	165.183456	42.962345	12.600389	-6.256737	1.305866
a_8	-243.449127	-79.080670	-31.360452	-11.786778	-19.701254
a_9	13.970771	4.851641	2.213764	0.165647	-3.627671
a_{10}	-693.537660	-219.992994	-69.773273	31.027071	155.927751

Table 7.2: Constants required to evaluate eq. (7.29) for different angles of attack

The variation of the regression coefficients for ΔC_d and $\Delta(l/d)$ found for different angle of attack, in most cases clearly show the effect of airfoil attitude on the aerodynamic effects of the step, however the erratic variation of coefficients for ΔC_m casts doubt over the accuracy of eq. (7.31).

Within the synthesis, the drag penalty of a non actuated step is found using eq. (7.29) in conjunction with a finite strip method. The airfoil shape at the midpoint of each strip is taken as the mean shape of the section. The overall change in lift and pitching moment are found using the same methodology, however the aerodynamic characteristics of the faired airfoil section must also be estimated using Xfoil. To account for any sweeping of the body/wing section, the local Mach number is corrected for sweep using:

$$M = M(y) \cos \Lambda_{c/4} \quad (7.32)$$

The resulting change of the aircraft's aerodynamic remaining coefficients, taken about the reference point (x_{ref}) is thus found as:

$$dC_D = \sum \left(\frac{dC_d}{h/c} \frac{h}{c} \right)_i \frac{S_i}{S_{ref}} \quad (7.33)$$

$$dC_L = \sum \left[\frac{d(L/D)}{h/c} \frac{dC_d}{h/c} \left(\frac{h}{c} \right)^2 \right]_i \frac{S_i}{S_{ref}} \quad (7.34)$$

$$dC_M = \sum \left[\frac{dC_m}{h/c} \frac{h}{c} + (0.75x_{le} + 0.25x_{te} - x_{ref}) \frac{d(L/D)}{h/c} \frac{dC_d}{h/c} \left(\frac{h}{c} \right)^2 \right]_i \frac{S_i c_i}{S_{ref} \bar{c}} \quad (7.35)$$

	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
R^2	0.961859	0.915771	0.902183	0.944320	0.974095
a_0	4592.497095	2502.209092	-4207.066836	-7679.35603	-8727.483884
a_1	-39.446976	-38.269159	-37.960036	-36.244206	-35.878079
a_2	-50569.99601	-64349.98299	-18441.81353	-17145.77381	-37983.60999
a_3	-16709.83519	-14536.36045	1693.156553	2928.348688	887.118334
a_4	324.846706	361.322155	180.497799	-188.983425	-314.786398
a_5	109752.2372	126428.5632	15328.96741	3103.336427	78500.57024
a_6	-397248.5808	-142497.387	177673.6993	645451.1701	1107787.073
a_7	309002.9443	172536.6381	-130577.4519	-149302.5426	-440876.1162
a_8	10983753.1	5296186.746	-185275.4382	-6549523.194	-15068233.13
a_9	48122.70107	34573.42973	19265.65818	37584.01695	17757.86293
a_{10}	-1970317.366	-1290721.089	-523092.0341	-1484375.939	-957181.2802

Table 7.3: Constants required to evaluate eq. (7.30) for different angles of attack

The sectional effects are found by interpolating the results given by equations for the angles of attack closest to the sectional angle of attack. The free stream angle of attack is used in all cases as local downwash velocities are not considered.

Based on the values of the coefficients of determination found for each regression and the errors between predicted and expected values, the drag equation (7.29) was deemed rather accurate and is routinely used in the synthesis. The lift (7.30) and moment (7.31) equations were found to produce less accurate results and given the time penalty associated with getting the sectional characteristics through Xfoil, are used in the synthesis only if the user so wishes.

	$\alpha = -5 \text{ deg.}$	$\alpha = -2 \text{ deg.}$	$\alpha = 1 \text{ deg.}$	$\alpha = 3 \text{ deg.}$	$\alpha = 6 \text{ deg.}$
R^2	0.895193	0.766023	0.780391	0.781331	0.881508
a_0	19.304458	24.898266	10.964602	9.821228	4.963287
a_1	-33.904348	-17.264869	-14.994449	-12.019588	-3.971041
a_2	-22.568818	-25.897159	-16.099015	-17.319254	-19.402382
a_3	-4.714347	-91.313019	6.211404	2.093671	-7.346682
a_4	104.166163	-1322.29326	-452.365596	-66.536699	639.369899
a_5	-25.07892	228.470519	4.173347	3.330552	16.682612
a_6	-1.045334	-1.125011	-0.964138	-0.800921	-0.37285
a_7	35.158461	13.028113	12.714447	20.071524	30.085079
a_8	-22.900827	-40.831503	-11.403347	-11.35638	-7.128946
a_9	-68.02433	291.807506	180.55995	57.151546	-243.064126
a_{10}	-32.634351	5.401606	4.641986	-10.328077	-16.542254

Table 7.4: Constants required to evaluate eq. (7.31) for different angles of attack

Chapter 8

Hydrostatics & dynamics

To produce a viable seaplane design, the hull chosen must exhibit a specific set of qualities. Low hydrodynamic resistance ensures that takeoff field length constraints are met and takeoff thrust requirements are minimised. The hull must also be able to plane at attitudes where it is not susceptible to the longitudinal hydrodynamic instabilities discussed in section 9.3.2. Lateral hydrostatic stability requirements must also be met by ensuring the wetted waterplane area width is sufficiently large.

This chapter presents the methods necessary to analyse a seaplane's waterborne performance and check that the design requirements are met. These methods are either established or novel but in all cases are based on the performance of past seaplane designs. Therefore, although the design of the BWB seaplane differs from traditional tube & wing flying boats in that the body extends outboard of the hull's chines, the hull's performance is assumed to remain unaltered.

8.1 Hull Sizing

In sizing the hull, the non dimensional hull parameters described in section 5.1 are mostly predefined at the start of the synthesis. The fuselage centreline length (L) is determined based on these parameters, along with the hull rise to beam fraction (ΔZ_{hr}) and hull beam loading at MTOW (C_{Δ_o}), in order for hull draft and performance requirements to be met.

When at rest, the hull must provide sufficient buoyancy to keep the passenger compartment at a sufficient height above the waterline. It must do so for the aircraft's maximum load (MTOW), even if a number of watertight compartments are damaged and fill with water. In the case of the BWB seaplane, the weight constraint is always considered, however given the aircraft's geometry it is assumed that there will always be sufficient reserve buoyancy for the cabin height constraints to be

met even if the watertightness of several compartments is compromised.

Based on Archimedes' principle, the weight (W) that can be kept afloat is proportional to the vessel's submerged volume (V).

$$W = Vgw \quad (8.1)$$

where w is the density of water and g the acceleration of gravity. In order for existing hull performance data to be applicable and the hydrodynamic resistance to remain unaffected, the body sections outboard of the hull rise must be kept above the waterline.

If the hull bottom shape is described by the function $z(x, y)$, obtained using the methods of section 5.4.1, the submerged volume of the hull for a given draft (d) is:

$$V = 2 \int_0^{y_{hr}} \int_{x_{le}(y)}^{x_{te}(y)} \left[Z'_l \left(\frac{t}{c} \right)_o L + d + (x - L_f) \tan \alpha - z(x, y) \right] dx dy \quad (8.2)$$

where the draft is defined as the vertical distance from the tip of the main step to the waterline and α is the equilibrium angle of attack when at rest on the water with respect to the aircraft baseline. The beam loading is therefore decreased and hull rise increment increased, as seen in figure 8.1, until:

$$V \geq \frac{W_o}{gw} \quad (8.3)$$

where the draft is given by:

$$d \leq (0.5 \tan \beta_1 + \Delta Z_{hr}) \sqrt[3]{\frac{W_o}{gw C_{\Delta_o}}} \quad (8.4)$$

There are however limitations to the values that these two parameters can take. The hull rise increment must be larger or equal to the main step height to beam ratio, A maximum value is also set to avoid the creation of less streamlined hulls and excessively thick hull segment airfoil sections. The beam loading is not constrained by a minimum value, however the minimum allowable beam length is governed by the preset spray severity constraint.

k_{sp}	Spray Characteristics
0.0525	Light
0.0675	Satisfactory
0.0825	Heavy but acceptable for overload
0.0975	Excessive

Table 8.1: Spray characteristics as determined by the empirical spray factor k_{sp}

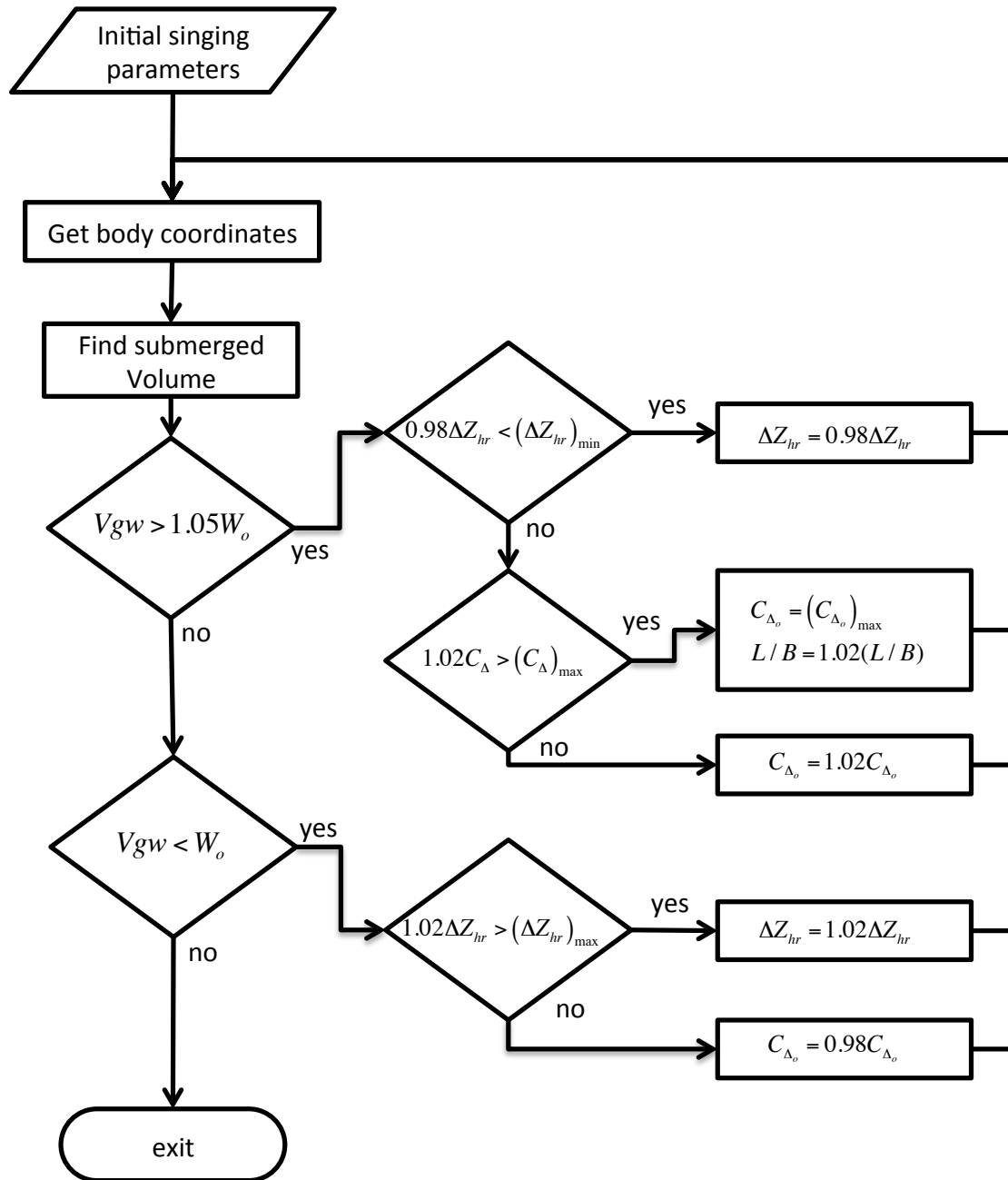


Figure 8.1: Flowchart illustrating the hull sizing process

The sizing requirements of a conventional flying boat hull forebody with respect to its spray intensity are empirically given by Parkinson [64] as:

$$C_{\Delta_o} = k_{sp} \left(\frac{L_f}{B} \right)^2 \quad (8.5)$$

The factor k_{sp} defines the hull's spray characteristics using the values given in table 8.1. The forebody length is governed by the location of the hull's main step, which in turn is determined by the aircraft's longitudinal centre of gravity position. Therefore for a given desired set of spray characteristics, the maximum beam loading must be given by:

$$(C_{\Delta_o})_{max} = k_{sp} \left(\frac{x_{cg}/L}{L/B} + \frac{x_{st}}{B} \right)^2 \quad (8.6)$$

where x_{cg} is the lengthwise position of the centre of gravity and x_{st} is the longitudinal distance between centre of gravity and the hull's main step, defined as positive when the CG is forward of the step.

If, when sizing the aircraft, the maximum beam loading constraint is reached and the hull rise increment has already been minimised yet the submerged volume remains larger than necessary, the predefined hull dimensions must be altered. By increasing the length to beam ratio, the forebody length to beam ratio is also increased. This move may lead to a linear increase in volume but also increases the maximum beam loading quadratically, allowing an overall reduction in volume.

A maximum value for k_{sp} between 0.0525 and 0.0725 can be chosen at the start of the synthesis. It is expected that the lighter the spray, the more applicable the methods described in section 8.3 will be due to the low additional resistance caused by spray wetting the part of the body intended to be out of the water.

8.2 Hydrostatic Analysis

Once the hull has been sized and the centre of gravity position identified, the aircraft's attitude on the water at different loading conditions can be calculated. When at rest, the only forces acting on a body partially submerged are hydrostatic pressure forces. Extending on eq. (8.2), the hydrostatic force (F_{hs}), pitching (M_{hs}) and rolling (L_{hs}) moments can be expressed in terms of the step draft, angle of attack and roll angle (γ).

$$F_{hs} = wg \int_{-b/2}^{b/2} \int_{x_{le}(y)}^{x_{te}(y)} \max(0, Z'_l(t/c)_o L + d + (x - L_f) \tan \alpha + y \tan \gamma - z(x, y)) dx dy \quad (8.7)$$

$$M_{hs} = wg \int_{-b/2}^{b/2} \int_{x_{le}(y)}^{x_{te}(y)} \max(0, Z'_l(t/c)_o L + d + (x - L_f) \tan \alpha + y \tan \gamma - z(x, y)) (x_{ref} - x) dx dy \quad (8.8)$$

$$L_{hs} = -wg \int_{-b/2}^{b/2} \int_{x_{le}(y)}^{x_{te}(y)} \max(0, Z'_l(t/c)_o L + d + (x - L_f) \tan \alpha + y \tan \gamma - z(x, y)) y dx dy \quad (8.9)$$

where the pitching moment is found with respect to the point x_{ref} and the rolling moment is calculated about the centreline. The equilibrium position can thus be found as the combination of d , α and γ that satisfy the following system of simultaneous equations.

$$\begin{cases} F - W = 0 \\ M - W(x_{ref} - x_{cg}) = 0 \\ L + W y_{cg} = 0 \end{cases} \quad (8.10)$$

The lines of a sample aircraft in it's equilibrium hydrostatic condition, obtained using the methods previously described, can be seen in figure 8.2.

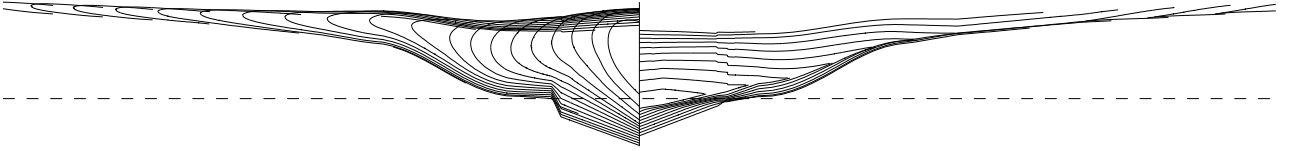


Figure 8.2: Hull lines and attitude of a seaplane at rest on water at MTOW and $\alpha = 3.59$ degrees

8.3 Hydrodynamic Analysis

A vessel moving through water experiences not only the hydrostatic pressure forces previously mentioned but also hydrodynamic pressure loads, resulting from the variation of flow velocity along the hull bottom. As hydrodynamic forces are proportional to the square of velocity and hydrostatic ones are a function of the submerged volume, one can see that as the velocity increases the vessel's weight becomes increasingly supported by the hydrodynamic forces, lifting it further out of the water and thus reducing the hydrostatic component. The vessel experiences resistance while moving in the water due to a combination of lift induced, frictional, spray and wave (not to be confused with the transonic type) resistance components.

A review of the available hydrodynamic analysis methods is given in section 3.4. Early on in the design process the ability to accurately and rapidly estimate the hydrodynamic performance of the vessel through the different speed ranges was identified as critical. The numerical methods for predicting the impact loading of an arbitrary hull section, discussed previously, were found to require excessive

computational effort. The strip theory based method presented by van Deyzen [95] was applied to a steady state problem and modified to use a semi-empirical relation that is better suited to hulls of arbitrary cross sections, based on an extension of the added mass method presented by Mayo [53]. When applied to flat transom planing hulls it showed great promise in predicting resistance loads but proved less accurate in predicting the hull's running attitude. This model also offered substantially greater flexibility in hull shapes that could be analysed than the empirical methodologies of Savitsky [78–80]. When applied to stepped hulls however, an accurate enough empirical formula to estimate the reattachment length aft of a step could not be found and attempts to generate one yielded mixed results.

Instead, given the wealth of experiment data available from NACA and ARC reports dating back to the 1920's, the creation of a new set of design methodologies was opted for. Resistance and pitching moment data for varying hull geometries, trim angles and loading conditions were taken from the reports referenced in appendix A, either from tables of experimental data or by digitising the graphs provided. Experimental data for the Series-62 [19] and Series-65 [40] prismatic hulls, featuring no afterbody structure, were also included in the database in the hope that the effects of the forebody alone could be better defined.

A major obstacle in further analysis proved to be, with the exception of some very early reports, the absence of experimental data on the vessel's instantaneous draft or wetted keel length, a parameter that is extensively used in previous empirical models to determine hydrodynamic lift and moments. Consequently the resistance and pitch moment could only be related to the hull's dimensions (seen in fig. 5.2), centre of gravity position, beam loading (C_Δ) and trim angle (τ). In all cases the trim angle is taken as the angle between the forebody keel at the step and the waterline.

Initially, the use of a neural network for the purpose of function approximation was attempted. However the number of nodes necessary to model the inherent nonlinearities of operating in the water-air interface in addition to the sheer number of data points (over 28,000) and prediction parameters proved excessive for the computational means available.

Instead, using the available data, a number of empirical models were generated. Due to the nonlinearity of the problem and the impossibility of manually identifying the effects of each parameter, a stepwise regression method was used. Stepwise regression, described in more detail in appendix D, uses successive multivariate least squares regressions to find the best fitting combination of predictor parameters and adds them to the model based on their statistical significance. A maximum of 15

descriptors was allowed to avoid overcomplicating the resulting empirical equations.

8.3.1 Parameter Choice

The parameters used to predict the resistance and pitching moment characteristics of a hull were not chosen arbitrarily but for their physical significance. As done for aerodynamic problems, dynamic similarity can be used to generalise experimental towing tank results for a specific hull shape. During takeoff and landing however seaplanes operate through two distinct regimes, the displacement regime at low speed and planing regime once hydrodynamic loading becomes dominant.

In the displacement regime, where viscous and wave making resistance are dominant, the displacement Froude number (4.23) and Reynolds numbers should be used to assess dynamic similarity. In scaling experimental results however Locke [50] uses only F_{∇}^2 , a practice which has been continued in this analysis as in the absence of data on the hull's draft the experimental wetted length Re values could not be accurately estimated.

$$F_{\nabla}^2 = \frac{C_v^2}{C_{\Delta}^{1/3}} \quad (8.11)$$

In addition to the previously defined geometric parameters, Savitsky and Brown [78] showed that in the displacement regime the parameter $\sqrt{2i_e}$ also has a substantial effect on the hull's wave making resistance. The hull's half angle of entrance (i_e) is the angle of the bow, seen in figure 8.3, at the hull's equilibrium waterline when at rest.

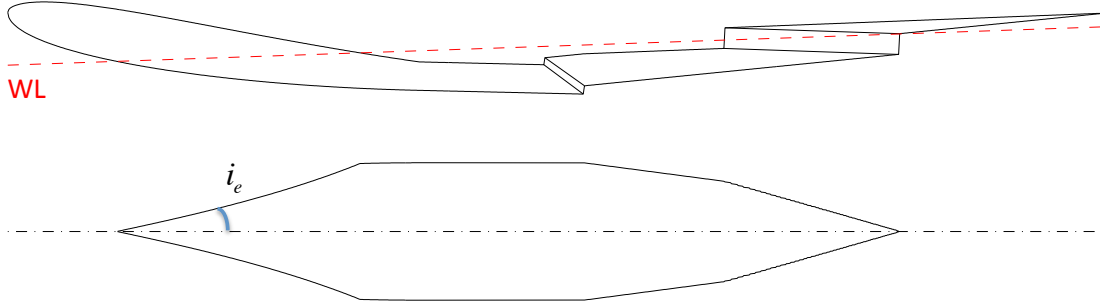


Figure 8.3: Definition of the hull's half angle of entry

In the planing regime Locke found that the hydrodynamic lift coefficient (8.12) is the similarity criterion for the flow about a planing hull, scaling both resistance and pitching moment values. Based on the available experimental results the vessel was found to be planing when the lift coefficient's square root lies in the following range:

$$0.3 \geq \sqrt{\frac{C_{\Delta}}{C_v^2}} > 0 \quad (8.12)$$

To provide adequate overlap of the low and high speed models, for $C_\Delta < 2$, the low speed model must therefore hold for:

$$1.5 < F_\nabla^2 < 15 \quad (8.13)$$

The hull's dimensional geometrical parameters are non-dimensionalised in terms of the vessel's beam to allow scaling of the experimental results. The parameters used for hulls with zero to two steps (N_{st}) therefore were:

$$\begin{aligned} L'_f &= L_f/B \\ \sec \beta'_1 &= \sec \beta_1 \\ \tan \delta'_{k_1} &= (\tan \delta_{k_1})^{(1.5N_{st}-0.5N_{st}^2)} \\ h'_1 &= \left(\frac{h_1/B}{\sin \delta_{s_1}} \right)^{(1.5N_{st}-0.5N_{st}^2)} \\ dL'_1 &= \left[(N_{st}-1) \left(\frac{L_2}{B} - \frac{L_f}{B} \right) - (N_{st}-2) \left(\frac{L}{B} - \frac{L_f}{B} \right) \right]^{(1.5N_{st}-0.5N_{st}^2)} \\ \sec \beta'_2 &= \left[\frac{(N_{st}-1)}{\cos \beta_2} - \frac{(N_{st}-2)}{\cos \beta_{ab}} \right]^{(1.5N_{st}-0.5N_{st}^2)} \\ \tan \delta'_{k_2} &= (\tan \delta_{k_2})^{(0.5N_{st}^2-0.5N_{st})} \\ h'_2 &= \left(\frac{h_2/B}{\sin \delta_{s_2}} \right)^{(0.5N_{st}^2-0.5N_{st})} \\ dL'_2 &= \left(\frac{L}{B} - \frac{L_2}{B} \right)^{(0.5N_{st}^2-0.5N_{st})} \\ \sec \beta'_3 &= (\sec \beta_{ab})^{(0.5N_{st}^2-0.5N_{st})} \end{aligned}$$

The secant of the various deadrise angles was used as it reflects the effect deadrise has on the hull wetted area and therefore viscous resistance. The same holds for the division of step height by the sine of the step's angle, as it reflects the step's face area.

Using these parameters along with the beam loading and trim angle, a set of predictors of the form

$$x_1^{p_1} x_2^{p_2} x_3^{p_3} \dots x_n^{p_n}$$

were generated, where the powers are predefined based on the physical behaviour of the problem. In generating the equations described in the following sections, over 3×10^6 distinct parameter combinations were defined.

The strict applicability range of all the equations present in the following sections can be seen in appendix E. Typically the equations can be used for approximately 20% beyond these values.

8.3.2 Resistance

The hull's resistance, taken parallel to the waterline, is typically presented in non-dimensional form as:

$$C_R = \frac{R}{\rho g w B^3} \quad (8.14)$$

As previously discussed, the way in which velocity affects the hull's resistance depends on whether the vessel is in the displacement or planing regimes. For the displacement regime ($1.5 < F_{\nabla}^2 < 15$), Locke [50] suggests that for a given trim angle, beam loading and hull geometry:

$$\frac{C_R}{C_v^2 C_{\Delta}^{2/3}} = f\left(\frac{C_v^2}{C_{\Delta}^{1/3}}\right) \quad (8.15)$$

where the nature of the problem requires that the function $f(F_{\nabla})$ is a power law with a negative exponent. The remaining parameters are assumed to vary with integer exponents from -4 to 4. The resulting empirical equation for resistance in the displacement range is given by eq. (8.16), which was found to have an R^2 of 0.9277 and a mean absolute percentage error of 9.86%.

$$\begin{aligned} \frac{C_R}{C_v^2 C_{\Delta}^{2/3}} = & a_0 + a_1 \frac{C_{\Delta}^{1/3}}{2i_e C_v^2} + a_2 \frac{C_{\Delta}}{L_f'^4} + a_3 \sqrt{\frac{C_{\Delta}^{1/3}}{C_v^2}} \left(\frac{L_f'^2 dL_2'^2 \sec^2 \beta_2' \sec^2 \beta_3' \tan \delta_{k_2}'}{\sec^2 \beta_1'} \right) + \\ & + a_4 \frac{C_{\Delta}^{4/3}}{C_v^2} \frac{L_f'^2 dL_1'^2 dL_2' \tan^2 \delta_{k_1} \sec \beta_2'}{\sec^2 \beta_1' \tan^2 \delta_{k_2}} + a_{10} \left(\frac{C_{\Delta}^{1/3}}{C_v^2} \right)^{3/4} \left(\frac{h_1' \sec \beta_3'}{L_f' \sec \beta_1'} \right)^2 + \\ & + a_5 \frac{C_{\Delta}^{1/3}}{C_v^2} \frac{\tan \tau (\sec \beta_1' \sec \beta_2' \sec \beta_3')^2 dL_1' dL_2'}{L_f'^2} + a_9 dL_2' \left(\frac{L_f' \tan \tau}{\sec \beta_1'} \right)^2 + \\ & + \frac{C_{\Delta}^{4/3}}{C_v^2} \left[a_6 \frac{\tan \tau L_f'^2 dL_2' \sec^2 \beta_1' \sec^2 \beta_3' \tan \delta_{k_1}'}{\sec^2 \beta_2' \tan^2 \delta_{k_2}'} + a_{12} h_2' \tan \tau \left(\frac{\sec \beta_1' \sec \beta_2'}{L_f' dL_1'} \right)^2 \right] \\ & + a_7 \left(\frac{C_{\Delta}^{1/3}}{C_v^2} \right)^{3/4} \frac{C_{\Delta} \tan \delta_{k_1}' \sec^2 \beta_2'}{L_f'^2 dL_1'^2 \sec^2 \beta_1' \sec^2 \beta_3' \tan^2 \delta_{k_2}'} + a_{11} \left(\frac{C_{\Delta}^{1/3}}{C_v^2} \right)^{1/4} + \\ & + a_8 dL_1' \tan \tau \left(\frac{L_f' \tan \delta_{k_1}'}{dL_2' C_{\Delta} \sec \beta_1' \sec \beta_2' \sec \beta_3' \tan \delta_{k_2}'} \right)^2 + \\ & + a_{13} \left(\frac{C_{\Delta}^{1/3}}{C_v^2} \right)^{3/4} C_{\Delta} h_2' \tan \tau \left(\frac{\sec \beta_2'}{L_f' dL_1' \sec \beta_1'} \right)^2 + a_{14} \frac{h_1' h_2' C_{\Delta}^{1/3}}{C_v^2 L_f'^2} \end{aligned} \quad (8.16)$$

For values of $F_{\nabla}^2 < 1.5$ the applicability of eq. (8.16) is doubtful. Therefore the resistance is assumed to vary with the square of velocity and is found in terms of the value predicted for $F_{\nabla}^2 = 1.5$ as:

$$C_R(F_{\nabla}^2) = \frac{C_v^2}{1.5 C_{\Delta}^{1/3}} C_R(1.5) \quad (8.17)$$

In the planing regime, the pressure loads and viscous resistance encountered are proportional to the square of the velocity, therefore:

$$\frac{\sqrt{C_R}}{C_v} = f\left(\frac{\sqrt{C_{\Delta}}}{C_v}\right) \quad (8.18)$$

The hull's resistance in the planning regime is consequently given by eq. (8.19). This equation was found to have a coefficient of determination $R^2 = 0.9798$ and a mean absolute percent error of approximately 5.57%.

$$\begin{aligned} \frac{\sqrt{C_R}}{C_v} = & a_0 + a_1 \frac{C_v}{\sqrt{C_{\Delta}} (\sec \beta'_1 \sec \beta'_2 \sec \beta'_3)^2} + a_2 \frac{C_v dL'_2 (L'_f dL'_1 \tan \delta'_{k_1} \sec \beta'_3)^2}{\sec \beta'_1} + \\ & + a_3 \frac{\tan \tau}{(\sec \beta'_1 \sec \beta'_2)^2} + \left[a_4 \frac{L'_f dL'_2 dL'_1{}^2}{(C_{\Delta} \sec \beta'_1 \sec \beta'_2)^2} + a_5 \sec \beta'_2 \right] \left(\frac{\sqrt{C_{\Delta}}}{C_v} \right)^{3/2} + \\ & + a_6 dL'_1 \tan \delta'_{k_1} \left(\frac{L'_f dL'_2 \tan \delta'_{k_2}}{\sec \beta'_1 \sec \beta'_2 \sec \beta'_3} \right)^2 \left(\frac{\sqrt{C_{\Delta}}}{C_v} \right)^{3/2} + \\ & + a_7 \frac{dL'_1 dL'_2}{C_{\Delta}} \left(\frac{L'_f}{C_v \sec \beta'_1 \sec \beta'_2 \sec \beta'_3} \right)^2 + a_8 \frac{L'_f C_{\Delta}}{C_v^2} \left(\frac{dL'_1 \tan \delta'_{k_1} \tan \delta'_{k_2} \sec \beta'_2}{\sec \beta'_1 \sec \beta'_3} \right)^2 + \\ & + a_9 \frac{\tan \tau}{\sec^2 \beta'_1} \left(\frac{\sqrt{C_{\Delta}}}{C_v} \right)^{3/2} + a_{10} \frac{C_v \tan \tau L'_f \tan \delta'_{k_1} h'_2}{\sqrt{C_{\Delta}} \sec^2 \beta'_2} + a_{15} \frac{C_v}{\sqrt{C_{\Delta}}} \left(\frac{L'_f}{h'_1 h'_2 \sec \beta'_1} \right)^2 + \\ & + \left(a_{11} + a_{13} \frac{\sqrt{C_{\Delta}}}{C_v} \right) \left(\frac{\tan \tau}{\sec \beta'_1 \sec \beta'_2} \right)^2 + \left(a_{12} + a_{14} \frac{C_{\Delta}}{C_v^2} \right) \left(\frac{\sec \beta'_1 \tan \tau}{\sec \beta'_2} \right)^2 + \end{aligned} \quad (8.19)$$

For combinations of C_v and C_{Δ} where both the displacement and planing regime applicability criteria are satisfied, the vessel is said to be near its hump speed, where resistance and trim angle are typically maximised. In this region, the resistance is simply predicted by taking a weighted average of the displacement and planing regime predictions. The same is done for the moment and trim expressions described in the next section.

a_0	-0.066897	a_5	-0.252521	a_{10}	-0.926827
a_1	-0.560225	a_6	-0.002939	a_{11}	0.158731
a_2	0.958649	a_7	0.123791	a_{12}	17.810755
a_3	-0.001457	a_8	-0.000444	a_{13}	-13.054648
a_4	0.000440	a_9	0.006440	a_{14}	1.217744

Table 8.2: Coefficients necessary to evaluate eq. (8.16) to obtain the displacement range resistance coefficient

a_0	0.043044	a_5	0.465130	a_{10}	0.003669
a_1	-0.000494	a_6	0.004258	a_{11}	1.881001
a_2	0.000044	a_7	-0.000371	a_{12}	-0.576745
a_3	-0.251041	a_8	-0.100026	a_{13}	3.341993
a_4	0.000090	a_9	-1.426938	a_{14}	0.426537
		a_{15}	1.2657×10^{-11}		

Table 8.3: Coefficients necessary to evaluate eq. (8.19) to obtain the planing range resistance coefficient

8.3.3 Pitching Moment

Being able to accurately predict the hull's running attitude is imperative, as the trim angle not only affects the amount of water resistance experienced by the vessel, but the designer must also ensure that it lies within the hydrodynamic stability bounds discussed in section 9.3.2. Generating accurate, simple equations to predict the hydrostatic and hydrodynamic pitching moment (M_H) of any hull proved less straightforward than in the case of C_R however. Early attempts to generate models for the hydrodynamic pitching moment coefficient C_{M_H} itself produced very unreliable expressions with very large mean absolute percent error values.

$$C_{M_H} = \frac{M_H}{wgB^4} \quad (8.20)$$

Instead two distinct methods were used to find either find the pitching moment in terms of the forces acting in the body (fig. 8.4) or the hull's equilibrium trim angle for a given applied pitching moment.

8.3.3.1 Centre of Pressure

The pitching moment is found in term's of the longitudinal position of the hull's centre of pressure, with respect to the hull's main step. This approach is advantageous as it inherently takes into consid-

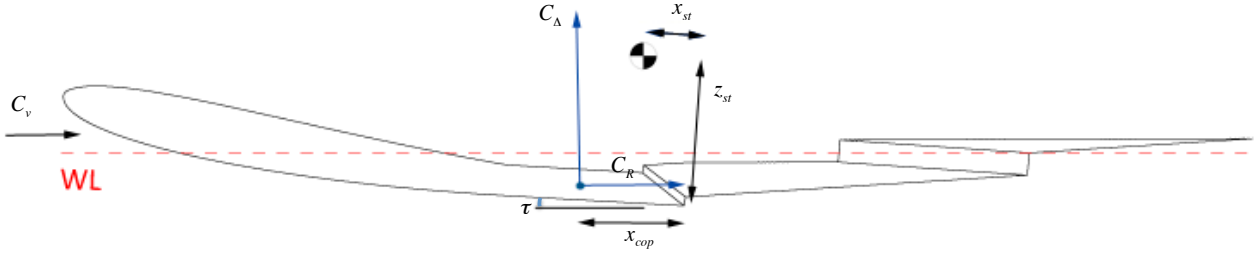


Figure 8.4: Forces acting on a seaplane hull

eration the location of the aircraft's centre of gravity, reducing by two the number of parameters that must be considered. The experimental centre of pressure position with respect to the hull's main step is calculated using:

$$\frac{x_{cop}}{B} = \frac{x_{st}}{B} + \frac{C_{MH}}{C_{\Delta} \cos \tau + C_R \sin \tau} \quad (8.21)$$

The nose down moment generated by the x-component of the hull's resistance force is neglected in this case, as the effect of pressure loads can be assumed to be dominant.

Using the regression parameters previously defined, the hull's centre of pressure in the displacement regime is given by eq. (8.22). A total of 13 regressors were used and the fit was found to be rather accurate with $R^2 = 0.8796$.

$$\begin{aligned} \frac{x_{cop}}{B} = & a_0 + a_1 \frac{dL'_2 (\sec \beta'_1 \sec \beta'_3)^2 C_{\Delta}^{1/3}}{(L'_f C_v)^2} + a_2 \sqrt{\frac{C_{\Delta}^{1/3}}{C_v^2} \frac{\tan \tau L'_f dL'_2}{\cos^2 \beta'_1 \cos \beta'_3}} + \\ & + a_3 C_{\Delta} h'_1 \left(\frac{dL'_2 \sec \beta'_3}{L'_f \sec \beta'_1} \right)^2 + a_4 \tan \tau \sec^2 \beta'_1 \sec \beta'_3 + \\ & + a_5 \sqrt{\frac{C_v^2}{C_{\Delta}^{1/3}} \frac{C_{\Delta} h'_1 \sec \beta'_1}{(dL'_2 \cos \beta'_3)^2}} + a_6 \frac{\tan \tau L'_f{}^2 dL'_1 C_{\Delta}^{1/3}}{(C_v \cos \beta'_2 \cos \beta'_2)^2} + \\ & + a_7 \left(\frac{\tan \tau L'_f dL'_1}{C_v C_{\Delta} \cos \beta'_1 \cos \beta'_2} \right)^2 C_{\Delta}^{1/3} + a_8 \frac{L'_f h'_2}{(L'_1 \cos \beta'_2)^2} + \\ & + a_9 \left(\frac{L'_f}{dL'_1 \cos \beta'_2} \right)^2 + a_{10} C_v C_{\Delta}^{5/6} dL'_1 \cos^2 \beta'_2 + a_{11} \frac{C_v^2 \tan \tau h'_2 \sec^2 \beta'_1}{(dL'_1 \sec \beta'_2)^2 C_{\Delta}^{1/3}} + \\ & + a_{12} \frac{C_v^2}{C_{\Delta}^{1/3}} \left(\frac{\tan \tau L'_f dL'_1 h'_2}{\cos \beta'_1 \cos \beta'_2} \right)^2 + a_{13} \left(\frac{C_v^2 L'_f}{C_{\Delta}^{1/3}} \right)^2 \sec \beta'_1 \end{aligned} \quad (8.22)$$

As for the resistance, for values of $F_{\nabla}^2 < 1.5$ the centre of pressure is found using:

$$x_{cop}(F_{\nabla}^2) = (x_{cop})_o + \frac{C_v^2}{1.5 C_{\Delta}^{1/3}} [x_{cop}(1.5) - (x_{cop})_o] \quad (8.23)$$

where $(x_{cop})_o$ is obtained by applying eq. (8.21) to the hydrostatic moment found using eq. (8.8) and $C_R = 0$.

a_0	1.085413	a_5	0.057709	a_{10}	0.022780
a_1	-4.027830	a_6	-0.102048	a_{11}	-0.305142
a_2	-0.120028	a_7	-0.044302	a_{12}	-0.001832
a_3	7.670381	a_8	0.149415	a_{13}	-0.002843
a_4	-2.665985	a_9	0.018263		

Table 8.4: Coefficients necessary to evaluate eq. (8.22) to obtain the displacement range centre of pressure position

For the planing regime model, the hydrodynamic lift coefficient was substituted for C_v^2 to increase the flexibility of the regression process. The resulting model, seen as eq. (8.24) uses 15 parameters and the fit quality is given by $R^2 = 0.8199$.

$$\begin{aligned}
\frac{x_{cop}}{B} = & a_0 + a_1 L'_f \left(\frac{C_\Delta h'_1 dL'_2 \sec \beta'_3}{C_v \sec \beta'_1} \right)^2 + a_2 h'_1 dL'_2 \left(\frac{\tan \tau L'_f}{C_v \sec \beta'_1 \sec \beta'_3} \right)^2 + \\
& + a_3 \frac{C_\Delta L'_f h'_1 dL'_2 \sec^2 \beta'_3}{C_v \sec^2 \beta'_1} + a_4 \frac{C_\Delta L'_f dL'_2 \sec^2 \beta'_3}{\sec^2 \beta'_1} + a_5 \frac{\tan \tau}{(\cos \beta'_1 \cos \beta'_3)^2} + \\
& + a_6 \frac{C_v dL'_2}{(\sec \beta'_1 \sec \beta'_3)^2} + a_7 C_v dL'_2 \left(\frac{L'_f \tan \tau}{\cos \beta'_1 \cos \beta'_3} \right)^2 + \\
& + a_8 dL'_2 \left(\frac{C_v L'_f}{C_\Delta \sec \beta'_1 \sec \beta'_3} \right)^2 + a_9 \left(\frac{C_v \sec \beta'_1 \sec \beta'_3}{L'_f dL'_2} \right)^2 + \\
& + a_{10} \frac{C_\Delta L'_f \sec^2 \beta'_2}{C_v \sec^2 \beta'_1} + a_{11} \frac{h'_2}{C_v} \left(\frac{\tan \tau L'_f dL'_1}{C_\Delta \cos \beta'_1 \cos \beta'_2} \right)^2 + \\
& + a_{12} h'_2 \tan \tau \left(\frac{\tan \delta'_{k_1} dL'_1}{C_\Delta \sec \beta'_1 \sec \beta'_2} \right)^2 + a_{13} \left(\frac{\tan \tau L'_f dL'_1}{\sec \beta'_1 \cos \beta'_2} \right)^2 + \\
& + a_{14} \left(\frac{\tan \tau L'_f dL'_1}{\cos \beta'_1 \cos \beta'_2} \right)^2 + a_{15} C_v \left(\frac{L'_f \sec \beta'_2}{h'_2 C_\Delta \sec \beta'_1} \right)^2
\end{aligned} \tag{8.24}$$

Looking at equations (8.22) and (8.24) the lack of parameters featuring the afterbody keel angles δ_{k_1} and δ_{k_2} is evident. This reinforces the feeling that the centre of pressure models, although able to model the cases in the rather expansive dataset supplied with reasonable accuracy, do not capture the physical behaviour of a hull moving through the water. In part this departure from physicality

a_0	0.960624	a_5	-3.150701	a_{10}	0.634422
a_1	-5.876483	a_6	-0.041067	a_{11}	-0.004206
a_2	-19.182322	a_7	-0.036683	a_{12}	0.078992
a_3	2.437516	a_8	-9.433688×10^{-7}	a_{13}	0.206438
a_4	0.016690	a_9	-0.008310	a_{14}	-0.210672
		a_{15}	1.009301×10^{-8}		

Table 8.5: Coefficients necessary to evaluate eq. (8.24) to obtain the planing range centre of pressure position

could be attributed to the initial decision to partially overlook the effect of resistance forces. Instead the centre of pressure longitudinal position can be calculated as:

$$\frac{x_{cop}}{B} = \frac{x_{st}}{B} + \frac{C_{MH} - \left(\frac{z_R - z_{st}}{B} \right) C_R \cos \tau}{C_\Delta \cos \tau + C_R \sin \tau} \quad (8.25)$$

where z_R is the vertical height above the keel at the main step of the point that the resistance tensor is assumed to act upon. This formulation was not used as the value of z_R could not be estimated accurately from the available experimental data, without knowledge of the hull's draft. An approximate value of

$$z_R = 0.25B \tan \beta_1$$

could be used, assuming that the chines of the hull near the step remain wetted, something that only holds at high beam loadings or low speed. Whether using this alternate relation would yield a more physically correct model however remains to be seen.

8.3.3.2 Equilibrium Trim Angle

If eqs. (8.22) and (8.24) are used in conjunction with eq. (8.26), the aircraft's equilibrium trim value can be found numerically by solving

$$C_{MH}(\tau, U_\infty) g w B^4 + 0.5 \rho U_\infty^2 \bar{c} S_{ref} C_M(\tau) - T(\bar{z}_{eng} - z_{cg}) = 0 \quad (8.26)$$

where T is the applied thrust and $\bar{z}_{eng}$ is the weighted mean thrust line height.

In producing the regression model, the hydrodynamic moment coefficient is used in the place of trim. In the displacement regime, a total of 16 regressors were used, producing an expression with $R^2 = 0.8110$ and a mean absolute percent error of 20%.

$$\begin{aligned}
\tau = & a_0 + a_1 \frac{C_\Delta^{1/6}}{2i_e C_v} + \sqrt{\frac{C_\Delta^{1/3}}{C_v^2}} \left(a_2 \frac{dL'_1 dL'_2 \tan \delta'_{k_1} \cos \beta'_1 \cos \beta'_2 x_{st} z_{st}}{C_\Delta L'_f \cos \beta'_3} + a_3 \frac{L'_f L'_2 x_{st}}{C_\Delta \cos \beta'_1 \cos \beta'_2} \right) + \\
& + a_4 \sqrt{\frac{C_\Delta^{1/3}}{C_v^2}} \frac{dL'_2 \tan^2 \delta'_{k_1} \cos \beta'_3 z_{st}}{L'_f \cos \beta'_1 dL'_1 \tan \delta'_{k_2} \cos \beta'_2} + a_5 \frac{C_v^2 C_\Delta^{2/3} \cos \beta'_1}{L'_f \cos \beta'_2 \cos \beta'_3} + \\
& + a_6 \frac{C_v^2 C_\Delta^{2/3} C_{MH} \cos \beta'_1 \cos \beta'_2 \tan \delta'_{k_2} x_{st} z_{st}}{L'_f dL'_2 \cos \beta'_3} + a_7 \frac{C_v^4 C_{MH} \cos \beta'_1 \cos \beta'_3 z_{st}}{C_\Delta^{5/3} L'_f dL'_1 \cos \beta'_2} + \\
& + \frac{C_v^4 C_{MH} x_{st}}{C_\Delta^{5/3}} \left(a_8 \frac{dL'_2 \cos \beta'_1}{L'_f \cos \beta'_2} + a_9 \frac{z_{st}}{dL'_1 dL'_2 \cos \beta'_1 \cos \beta'_3} \right) + a_{10} \frac{C_v^6 \cos \beta'_2}{L'_f \cos \beta'_1 \cos \beta'_3} + \\
& + a_{11} \sqrt{\frac{C_\Delta^{1/3}}{C_v^2}} \frac{C_\Delta dL'_2 x_{st}}{h'_1 L'_f \cos \beta'_1 \cos \beta'_3} + a_{12} \frac{C_v^4 h'_2 x_{st}}{L'_f dL'_1 \cos \beta'_1} + a_{13} \frac{C_v^4 C_\Delta^{1/3} \cos \beta'_1}{L'_f} + \\
& + a_{14} \frac{C_v^6 h'_2 x_{cg}}{C_\Delta L'_f \cos \beta'_1 dL'_1 \cos \beta'_2} + a_{15} \frac{h'_2 x_{st} \cos \beta'_1}{L'_f h'_1} + a_{16} \frac{C_\Delta dL'_1{}^2 x_{st}}{\cos^2 \beta'_3}
\end{aligned} \tag{8.27}$$

Again the trim for $F_\nabla^2 < 1.5$ is found by interpolating between the equilibrium trim at rest and the value obtains by evaluating eq. (8.27) at $F_\nabla^2 = 1.5$

a_0	8.813970	a_6	-37.591150	a_{12}	-0.239944
a_1	-137.827110	a_7	-21.118556	a_{13}	-0.388845
a_2	-12.969465	a_8	-14.251464	a_{14}	0.0129009
a_3	-0.370024	a_9	4.514150	a_{15}	0.164791
a_4	77.124105	a_{10}	-0.021662	a_{16}	-0.263701
a_5	0.180602	a_{11}	-0.453808		

Table 8.6: Coefficients necessary to evaluate eq. (8.27) to obtain the equilibrium trim in the displacement speed range

The hull's attitude in the planning regime is given by eq. (8.28), where using 14 terms the model was found to have an $R^2 = 0.7625$ and a mean absolute percent error around 22%.

a_0	4.312144	a_5	-6.775394	a_{10}	-80.639190
a_1	-0.604255	a_6	738.298999	a_{11}	-25.980955
a_2	-0.259172	a_7	-5.340073	a_{12}	1164.422700
a_3	-26.297563	a_8	0.739249	a_{13}	0.053739
a_4	-762.416940	a_9	219.231720	a_{14}	-9.729662

Table 8.7: Coefficients necessary to evaluate eq. (8.28) to obtain the equilibrium trim in the planing speed range

$$\begin{aligned}
\tau = & a_0 + \sqrt{\frac{C_v}{\sqrt{C_\Delta}}} \left(a_1 \frac{dL'_2 x_{st}}{\sec \beta'_1 \sec \beta'_2 \sec \beta'_3} + a_2 \frac{C_\Delta dL'_1 \cos \beta'_3 z_{st}}{L'_f \tan \delta'_{k_2} \cos \beta'_1 \cos \beta'_2} + a_3 \frac{C_{M_H} \cos \beta'_1 \cos \beta'_2}{L'_f dL'_1 \cos \beta'_3} + \right. \\
& + a_4 \frac{C_{M_H}^2 \tan^2 \delta'_{k_1} \tan \delta'_{k_2} \cos \beta'_1 \cos \beta'_2 x_{st}}{C_\Delta L'_f dL'_1 dL'_2 \cos \beta'_3} \left. \right) + a_5 \frac{C_{M_H} \cos \beta'_1 \cos \beta'_2}{C_\Delta L'_f \cos \beta'_3} + \\
& + \frac{\sqrt{C_\Delta}}{C_v} \left(a_6 \frac{C_\Delta dL'_1 \cos \beta'_1 \cos \beta'_3}{L'_f \cos \beta'_2} + a_7 \frac{C_\Delta}{L'_f \cos \beta'_1 \cos \beta'_2 \cos \beta'_3} + a_8 \frac{C_\Delta L'_f \cos \beta'_3}{\tan \delta'_{k_2} \cos \beta'_1 \cos \beta'_2} + \right. \\
& + a_{14} h'_2 dL'_1{}^2 \tan \delta'_{k_1} x_{xt} z_{st} \left. \right) + a_9 \frac{C_\Delta \cos \beta'_1 \cos \beta'_2 z_{st}}{C_v^2 L'_f : dL'_1} + \\
& + \left(\frac{\sqrt{C_\Delta}}{C_v} \right)^3 \left(a_{10} \frac{dL'_2 \sec \beta'_1 \sec \beta'_2 \sec \beta'_3}{C_\Delta L'_f} + a_{11} \frac{dL'_2 \cos \beta'_2 x_{st} z_{st}}{C_\Delta \cos \beta'_1 \cos \beta'_2} + \right. \\
& + a_{12} \frac{C_{M_H} dL'_2 \cos \beta'_1 \cos \beta'_2 \tan \delta'_{k_2}}{L'_f dL'_1 \cos \beta'_3} \left. \right) + a_{13} \frac{L'_f x'_{st} \cos \beta'_1 \cos \beta'_2}{dL'_1 h'_2}
\end{aligned} \tag{8.28}$$

In using eqs. (8.27) and (8.28) the hydrodynamic pitching moment necessary to balance the moments imparted by the aircraft's aerodynamic and propulsive forces is given by:

$$C_{M_H} = - \frac{0.5 \rho U_\infty^2 \bar{c} S_{ref} C_M(\tau) - T(\bar{z}_{eng} - z_{cg})}{g w B^4} \tag{8.29}$$

8.4 Validation

To validate the proposed method intended for evaluating the hull's hydrodynamic qualities, the NACA 161A-1 and 165A-1 tank test models [46] were chosen, as the source presented the most complete set of aircraft aerodynamic characteristics found. Using these aircraft, an unbiased validation could be obtained, as these experimental results were not used in building the model. Moreover, using experimental data from a full aircraft model rather than a lone hull section, the accuracy of the pro-

posed methodology in predicting the equilibrium trim angle and resistance for varying aerodynamic pitching moments and lift forces could be investigated.

Both these models feature a twin step configuration, however the second step location could also be considered as the end of the hull. The validation tests were therefore repeated for both the twin step configuration and the equivalent single step hull. The hull shape parameters and aerodynamic characteristics of the two hull, for both a single and dual step formulation can be seen in tables 8.8 and 8.9. In all cases the centre of gravity is at 28% of the mean aerodynamic chord.

	161A-1		165A-1	
	$N_{st} = 1$	$N_{st} = 2$	$N_{st} = 1$	$N_{st} = 2$
L (m)	2.492	3.543	2.568	3.324
C_{Δ_o}	0.619		1.044	
L/B	5.230	7.436	6.420	8.310
L_f/B	2.080		3.830	
L_{flat}/B	1.6		1.870	
β_1 (deg.)	20		22.5	
δ_{k_1} (deg.)	6.3		7.25	
h_1/B	0.107		0.079	
δ_{s_1} (deg.)	90.0		60.0	
L_2/B	-	5.320	-	6.420
δ_{k_2} (deg.)	-	16.0	-	10.0
h_2/B	-	0.120	-	0.310
δ_{s_2} (deg.)	-	17.0	-	12.5
β_2 (deg.)	-	20.0	-	22.5
β_{ab} (deg.)	20.0	5.0	22.5	5.0

Table 8.8: Hull shape parameters of the validation models

Early on, the results of eq. (8.27) were found to be extremely erratic and not correspond to reality. Therefore eq. (8.22) was always used to predict the equilibrium trim angle for $1.5 < F_{\nabla} < 15$. In the hump region, where $F_{\nabla} \leq 15$ and $\sqrt{C_{\Delta}}/C_v \leq 0.3$, the trim value was found as a combination of the

	$\delta_e = 0^\circ$	$\delta_e = -25^\circ$
C_{L_o}	0.364	0.313
$dC_L/d\tau$	5.655	5.203
$C_{L_{\max}}$	1.635	1.577
C_{M_o}	0.040	0.337
$dC_M/d\tau$	-1.149	-1.326
$S_{ref} (m^2)$		2.584
$\mathcal{R}$		7.1
CM_q		-0.125

Table 8.9: Aerodynamic characteristics of the validation models referenced to $0.24\bar{c}$

high and low speed results using:

$$\tau_{LS} + (\tau_{HS} - \tau_{LS}) \left(\frac{C_v - \sqrt{C_\Delta}/0.3}{\sqrt{15C_\Delta}^{1/3} - \sqrt{C_\Delta}/0.3} \right) \quad (8.30)$$

The same formula can be used for the hump resistance by substituting τ with C_R . The computed and experimental free to trim paths, with no power applied and for varying elevator deflections can be seen in figs. 8.5, 8.6, 8.9 and 8.10 for the equivalent single step hull and in figs. 8.7, 8.8, 8.11 and 8.12 for the twin-step hull.

Looking at these plots, one sees that both models accurately capture the general behaviour of the model in most cases. The resistance in the displacement range closely resembles the experimental one, whenever the predicted trim angle is correct. This is the case for both single step hulls, however for the twin step 161A-1 hull, the equilibrium trim angle is severely underestimated in the displacement regime, affecting the predicted resistance values.

Both the methods shown rather accurately predict the hump velocity, the speed at which resistance and running attitude are maximised, the centre of pressure formulation exhibiting a higher level of accuracy in most cases. However it is clearly evident that equilibrium trim is underestimated at the hump speed and in the subsequent planing regime, which results in an underestimation of the hull resistance force.

It should however be noted that an increasing under prediction of total resistance should be expected as speed increases, as the various aerodynamic drag components were not given by the source and were thus not included in the computed resistance value. The equations seen in section 8.3.2 only

model the hydrodynamic resistance component and therefore the aerodynamic drag must be added based on the shape and aerodynamic performance of the aircraft in question. Had an accurate value for aerodynamic skin friction, pressure and induced drag been provided for subsequent use in the comparison, the perceived difference between the experimental and predicted high speed resistance values would have been substantially lower.

Results obtained using the centre of pressure method were found to best relate to experimental results, compared to those given by the equilibrium trim regression. This is particularly the case for single step hulls. The need to iteratively find the equilibrium trim angle however does have an associated computational cost which should be considered when choosing to use this formulation.

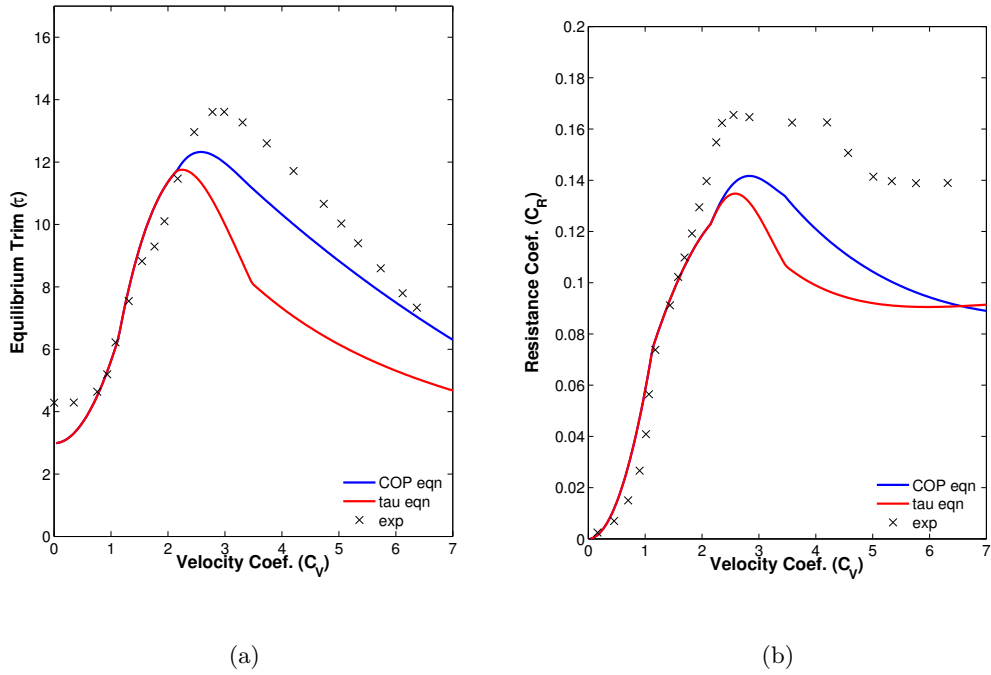
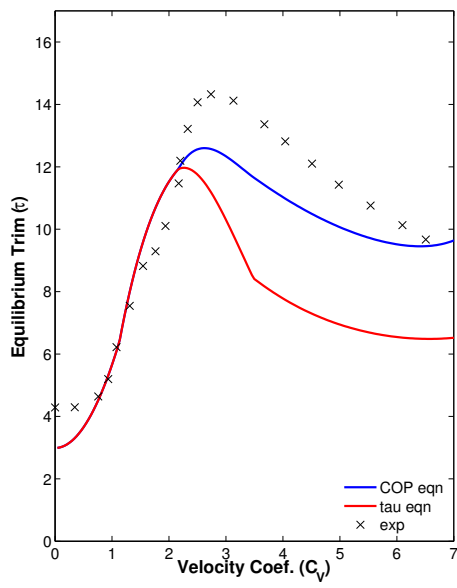
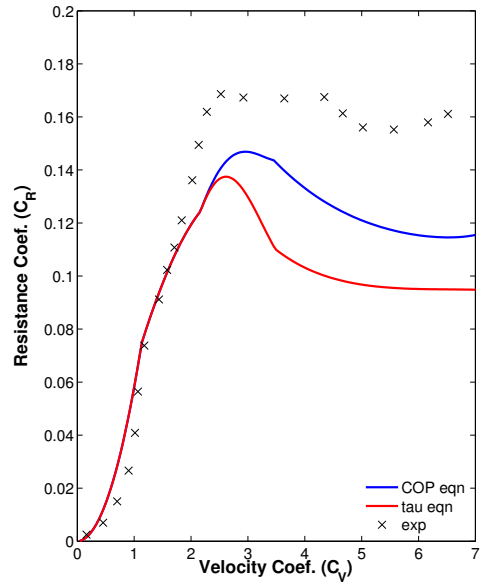


Figure 8.5: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the single step 161A-1 model with $\delta_e = 0^\circ$

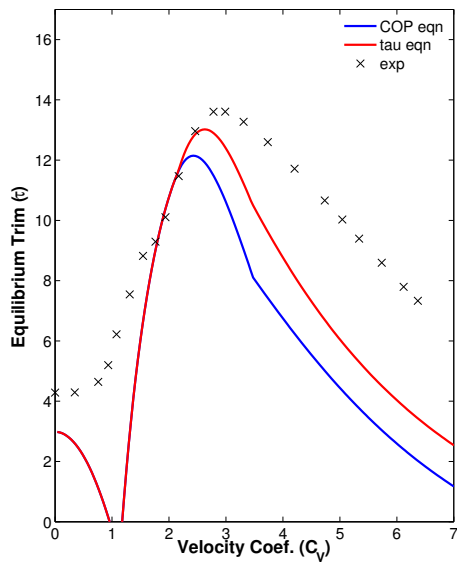


(a)

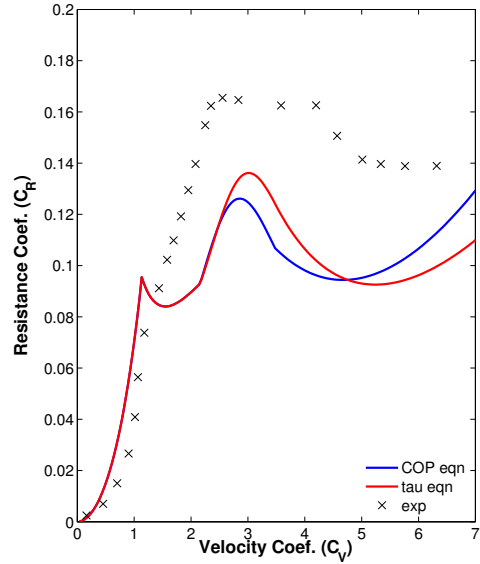


(b)

Figure 8.6: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the single step 161A-1 model with $\delta_e = -25^\circ$

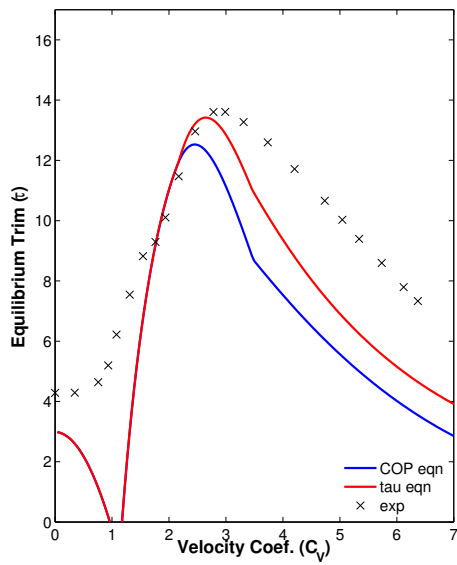


(a)

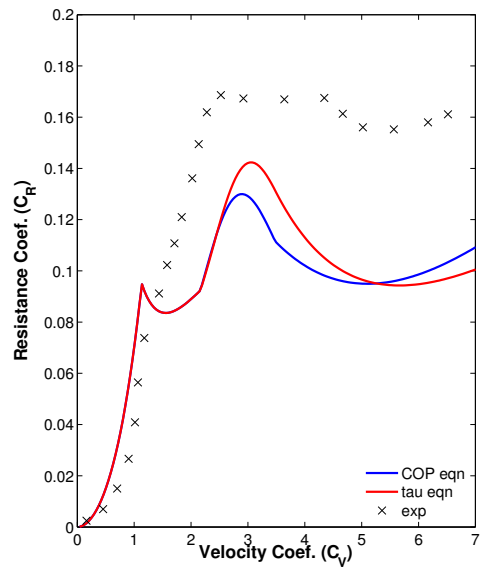


(b)

Figure 8.7: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the twin step 161A-1 model with $\delta_e = 0^\circ$

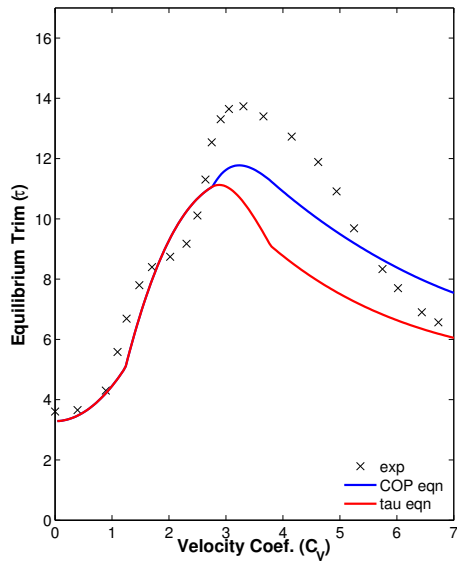


(a)

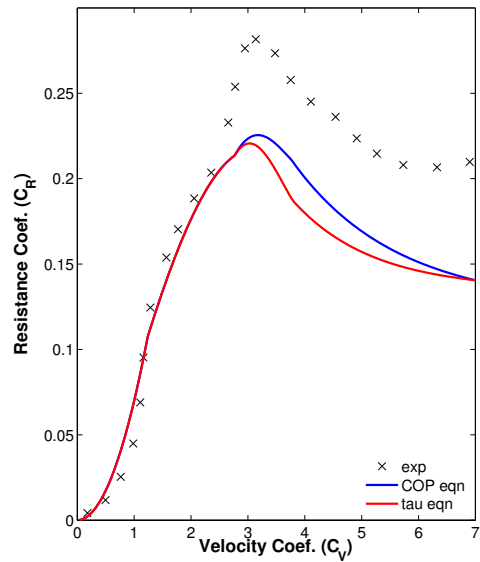


(b)

Figure 8.8: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the twin step 161A-1 model with $\delta_e = -25^\circ$

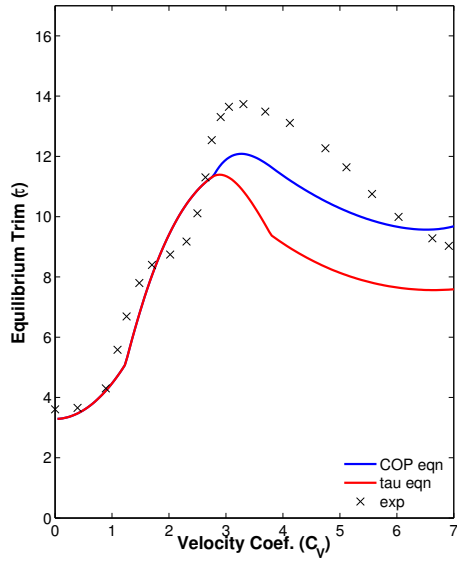


(a)

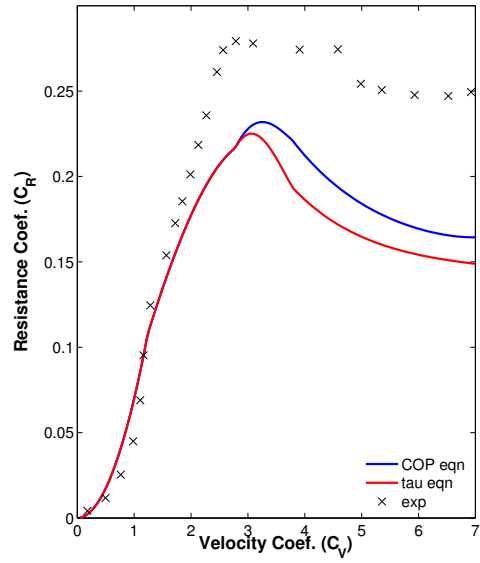


(b)

Figure 8.9: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the single step 165A-1 model with $\delta_e = 0^\circ$

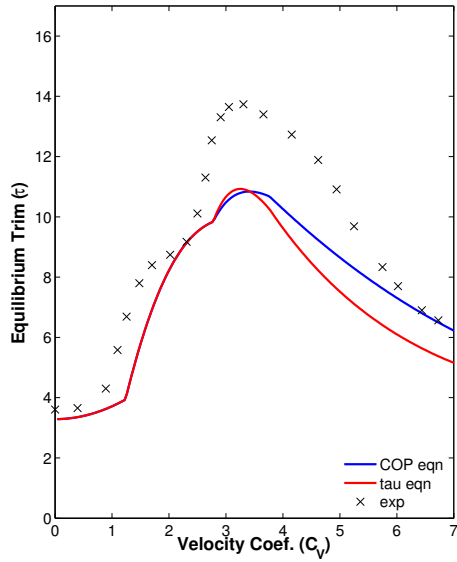


(a)

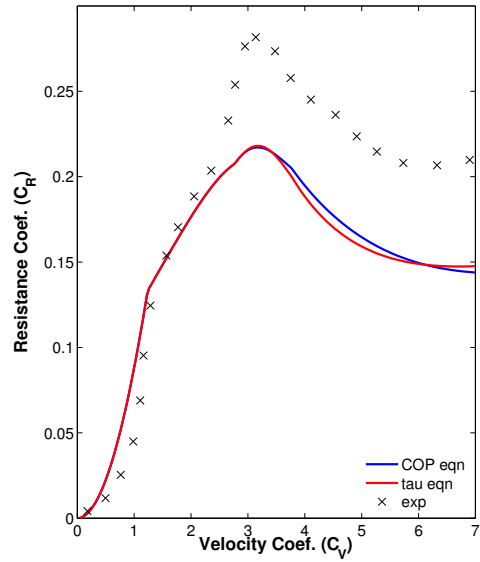


(b)

Figure 8.10: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the single step 165A-1 model with $\delta_e = -25^\circ$

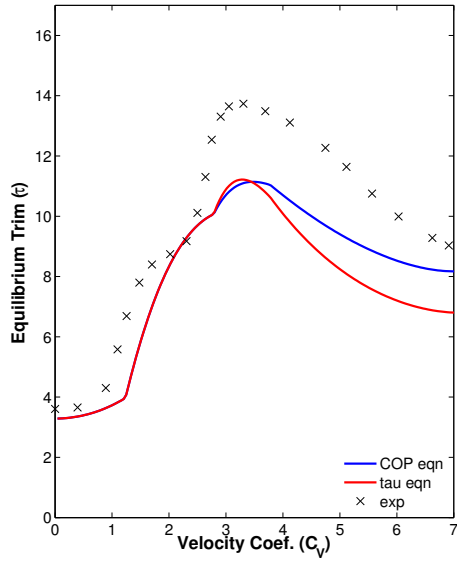


(a)

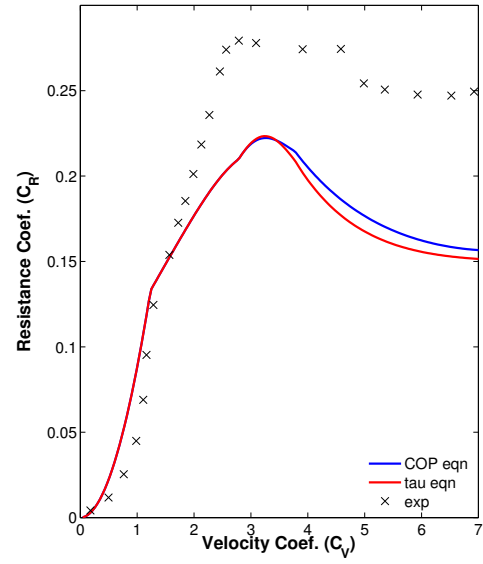


(b)

Figure 8.11: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the twin step 165A-1 model with $\delta_e = 0^\circ$



(a)



(b)

Figure 8.12: Plot of equilibrium trim angle (a) and resistance (b) vs. Velocity coefficient for the twin step 165A-1 model with $\delta_e = -25^\circ$

Chapter 9

Weight, Balance & Stability

The aircraft's static stability, i.e. its ability to return to its original equilibrium position following a perturbation, is dependent on the aircraft's aerodynamic and mass characteristics. This chapter addresses the methods used to ensure that the resulting aircraft has the desired levels of static stability both in flight and on the water. Attaining a statically stable hull or wing planform is heavily dependent upon the vertical and longitudinal position of the aircraft's centre of gravity (CG).

The aircraft's CG is found by considering the weight and location of individual system groups. To reflect standard practice and simplify calculation, the aircraft structure and payload are assumed to be symmetrical about the x-z plane. Also addressed in this chapter are the dynamic instabilities that seaplanes are susceptible to both in the air and on the water.

9.1 Weight & Balance

During each iteration of the design synthesis, the aircraft's weight and balance must be estimated. The methodology employed is split into two constituent parts. The estimation of the aircraft's Operating Empty Weight (OEW) and balance, which is considered a constant, and the effect of different loading conditions upon those characteristics.

9.1.1 Empty Aircraft

The aircraft's OEW accounts for the weight of all structural components and aircraft systems as well as the basic crew and all operational equipment necessary for the aircraft to complete its mission. Having an accurate prediction for the empty weight is necessary as the only value available to this point in the preliminary design process was the initial estimate found based on a rather crude empirical relation.

The OEW is found by summing all the individual component weights, calculated using a number of established empirical relations. Each individual relation to be used is chosen based on its applicability to this particular aircraft configuration and its intended mission. A full list of the methods used can be found in appendix H. The total weight and centre of gravity position (x_{cg}, z_{cg}) is subsequently found using:

$$W = \sum W_i \quad (9.1)$$

$$x_{cg} = \frac{\sum W_i x_i}{\sum W_i} \quad (9.2)$$

$$z_{cg} = \frac{\sum W_i z_i}{\sum W_i} \quad (9.3)$$

where W_i is the weight of the individual component and (x_i, y_i, z_i) is the component's centre of gravity position. The inertial properties of the aircraft can also be found using the parallel axis theorem in conjunction with each component's inertial properties, taken about its own centre of gravity.

$$I_{xx} = \sum \{ (I_{xx})_i + W_i [(z_i - z_{cg})^2 + y_i^2] \} \quad (9.4)$$

$$I_{yy} = \sum \{ (I_{yy})_i + W_i [(z_i - z_{cg})^2 + (x_i - x_{cg})^2] \} \quad (9.5)$$

$$I_{zz} = \sum \{ (I_{zz})_i + W_i [(x_i - x_{cg})^2 + y_i^2] \} \quad (9.6)$$

$$I_{xz} = \sum \{ (I_{xz})_i + W_i (z_i - z_{cg})(x_i - x_{cg}) \} \quad (9.7)$$

For structural components whose size or mass is substantially smaller than that of the entire aircraft, the components own inertial properties can be assumed to be negligible. Moreover given the assumption that the aircraft is symmetrical about the x-z plane, $y_{cg} = 0$ and the y-related products of inertia I_{zy} and I_{xy} are also zero.

A major departure from the standard set of relations used for conventional aircraft is the weight estimation of the aircraft's main structure. The structure is split into two distinct components. The outboard wing ($y > y_{le}$), whose structural weight is estimated using the methods presented by Torenbeek [93], and the wing centre section.

The methodology used to estimate the weight of the outboard wing, accounts not only for the structural weight necessary to deal with the forces that the aircraft is subjected to in flight but also the incremental weights of any spoilers and leading or trailing edge devices present on the wing. As the outer wing section will often be of non-trapezoidal planform, the equivalent reference wing of the section is used. The errors resulting from this assumption can be considered rather low as the methodology has previously proven accurate in estimating the weight of cranked wings, commonly used on modern airliners.

The section inboard of the outer wing section ($y < y_{le}$), housing the cabin and cargo bays cannot be structurally considered a wing and therefore cannot be analysed as one. Instead the empirical methodology developed by Bradley [13] was used. Based on the results of finite element computations and corrected for the values quoted by Boeing for their own designs, the method relates the cabin section weight to the maximum cabin floor area (S_{cab}) and MTOW.

$$W_{cab} = 1.8032W_o^{0.1666}S_{cab}^{1.0612} \quad (9.8)$$

,where W_o and S_{cab} are both in imperial units. The structural section aft of the cabin pressure bulkhead is considered to be structurally similar to a tailplane carrying engines and is thus treated as one.

$$W_{aft} = 0.53(1 + 0.05N_{eng})S_{aft}W_o^{0.2}(\lambda_{aft} + 0.5) \quad (9.9)$$

where λ_{aft} is the aft section's equivalent wing taper ratio. However accurate this methodology may be, it does not take into consideration the weight penalties of having to withstand severe waterborne landing impact loads.

Early in the design process the need for a methodology to estimate the weight penalty of waterborne operations was identified. Ideally the weight must be related to the fuselage dimensions and its approach speed, a measure of the impact loads experienced when landing on water.

In the absence of an existing empirical methodology and information on the fuselage weight of an adequately large sample of past designs, attempts were made to generate one by subtracting the weight of individual components from the quoted empty weight of numerous past seaplane and amphibian aircraft. The resulting fuselage weights were found to be substantially higher than those of the few aircraft that fuselage weight information were available for. The discrepancy is most likely as result of using methods intended for modern aircraft, on designs from the 1930-40's. There is also doubt whether

the resulting methodology would be applicable to a modern airliner, given the vast improvements that have been achieved in minimising an aircraft's structural weight over the past decades.

Consequently a very crude approach was taken, by adding a weight penalty equal to 60% of the predicted weight for a fuselage matching the hull's dimensions. The 1.6 penalty factor used is based on the suggestion made by Raymer [67]. The hull weight was thus found based on a modified version of the Class II empirical fuselage weight estimation formula given by Torenbeek [93] as:

$$W_{hull} = 0.006172 \sqrt{\frac{1.94V_D}{\left(\frac{t}{c}\right)_o + \frac{1}{L/B}}} \left\{ 1.8 \left[\left(\frac{t}{c}\right)_o + \frac{1}{L/B} \right] L^2 \right\}^{1.2} \quad (9.10)$$

where V_D is the design dive speed in m/s and the resulting weight is in kg .

The centre of gravity position of the cabin section and hull are likely to have a major effect on the aircraft's centre of gravity due the substantial weight contribution of this section. Lacking any information on the structural design of past seaplanes, the centre of gravity location cannot be estimated based on past experience, as is done in most other cases. Instead, the weight distribution along the cabin section is based on the relative severity of the impact loads likely to be experienced at each particular location. Airworthiness directives (Title 14 - CFR 25.527) require the the hull be able to withstand varying impact load factors along it's length. For a step landing the impact load factor (n_w) is given as:

$$n_w(L_f) = \frac{C_1 V_{S_L}}{W_L^{1/3} \tan^{2/3} \beta_1} \quad (9.11)$$

while for a bow or stern landing case

$$n_w(x) = \frac{C_1 V_{S_L}}{W_L^{1/3} \tan^{2/3} \beta(x)} \frac{K_1(x)}{\sqrt{1+r^2}} \quad (9.12)$$

where all values are in imperial units, V_{S_L} is the aircraft's stall velocity in the landing configuration, $C_1 \approx 0.012$, β is the local deadrise angle and r is a ratio of longitudinal distance from hull's centre of gravity to the location in question divided by the hull's radius of gyration in pitch. As both the centre of gravity and radius of gyration are unknown the value of r is simply approximated as the percentage length distance from the main step. The factor K_1 is an empirical hull station weighing factor in the present case approximated as:

$$K_1 = \begin{cases} 1.5 - 0.5 \frac{x}{L_f - L_{flat}} & , \text{ if } x < L_f - L_{flat} \\ 1 & , \text{ if } L_f - L_{flat} \leq x \leq L_f \\ 0.375 + 0.625 \frac{x - L_f}{L - L_f} & , \text{ if } L_f < x \leq L \end{cases} \quad (9.13)$$

Therefore assuming a bottom penalty factor at the main step (p_o) of approximately 130%, chosen based on the relative weight of the bottom to the remaining hull structure, the hull bottom surface weight penalty factor is found as:

$$\frac{p}{p_o} = \frac{n_w(x)}{n_w(L_f)} \approx \left(\frac{\tan \beta_1}{\tan \beta(x)} \right)^{2/3} \frac{K_1(x)}{\sqrt{1 + \left(\frac{x - L_f}{L} \right)^2}} \quad (9.14)$$

For the remainder of the cabin section, the top surface is assumed to have $p = 1$ and the bottom seawing surface $p \approx 1.05$. Sections where heavy frames, used to support the wing structure, are located are also penalised by approximately 15%. These relative values can be easily changed when more accurate information is available.

The CG position and inertial properties of this structural section are therefore found using the penalty factors defined above and numerically integrating the values found throughout the section.

9.1.2 Fuel & Payload

The limits to which the centre of gravity can travel during flight must be determined to ensure static stability under all circumstances. The conditions examined, other than a fully empty aircraft, are:

- No payload, fuel tanks filled forward to aft and aft to forward
- Normal payload distribution, no fuel or fuel tanks filled forward to aft and aft to forward
- Aftmost payload CG, no fuel or fuel tanks filled forward to aft
- Foremost payload CG, no fuel or fuel tanks filled aft to forward

The extreme payload distribution scenarios are considered to model the abrupt movement of passengers forward or aft in case of an emergency. As in such a case, passengers would also crowd the cabin's aisles, a maximum floor area of $0.25m^2$ is allocated per passenger. These cases are computed iteratively by filling the cabin up to the previously found position of the aft most/foremost CG.

The fuel tanks are filled in the predefined order, based on each fuel tank's longitudinal centre of gravity position. In case a fuel tank is only partially filled, the vertical and lateral centres of gravity are corrected for the fact that the fuel pools in the lowest available free space, however the effects of angle of attack are not considered.

Figure 9.1 shows the longitudinal centre of gravity envelope for a sample aircraft with forward, aft and wing fuel tanks in both empty and full payload configurations.

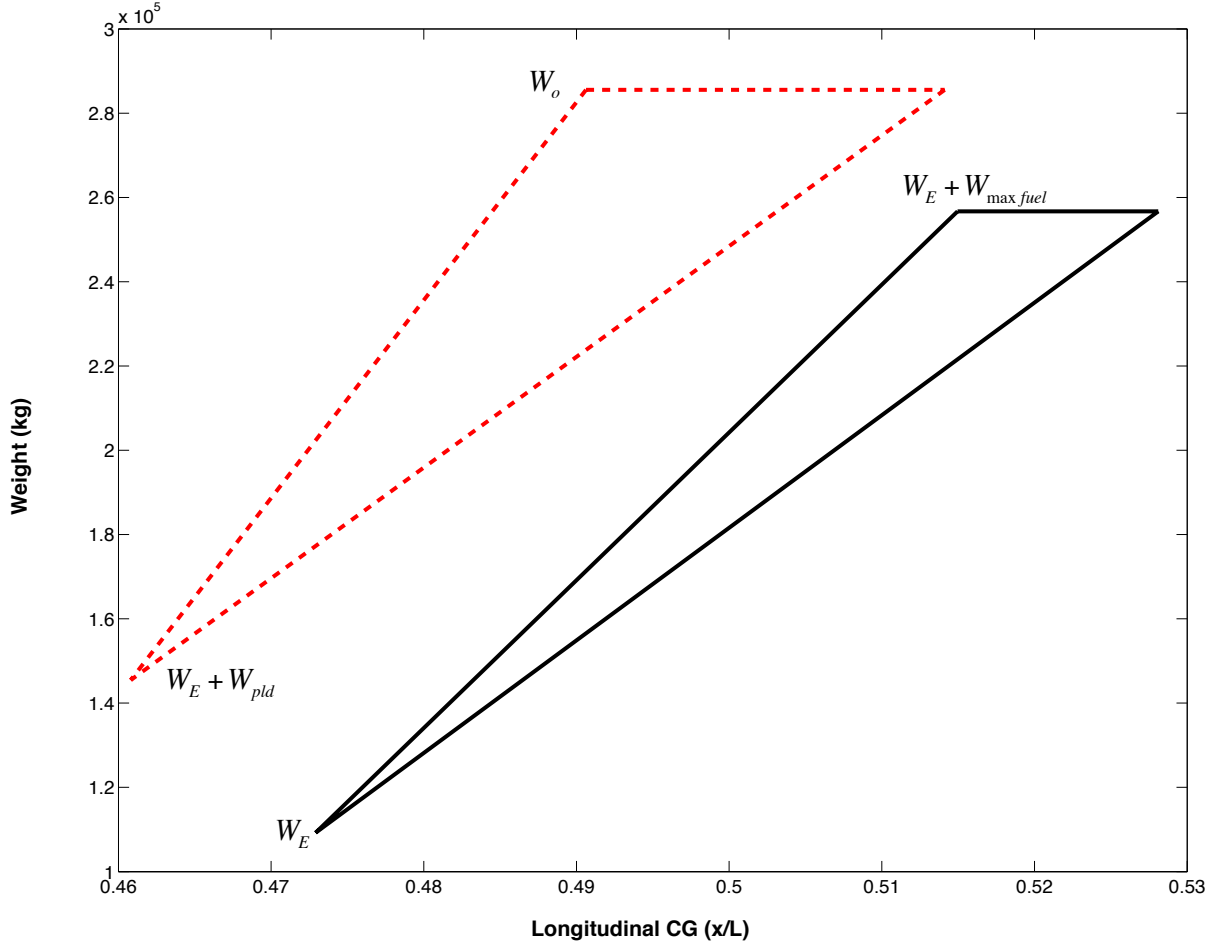


Figure 9.1: Centre of Gravity envelope for a sample aircraft

9.2 Static Stability

9.2.1 Aerodynamic

An aircraft's longitudinal static stability and therefore its response to a perturbation in pitch is characterised by its static margin. The static margin, which is defined as the distance between the aircraft's neutral point and its longitudinal CG position, must be positive for the aircraft to be statically stable. For an all-wing aircraft with no horizontal stabilisers, the neutral point is coincident with the wing's centre of pressure. The static margin can therefore be found using:

$$\frac{x_{sm}}{\bar{c}} = \frac{x_{np} - x_{cg}}{\bar{c}} = -\frac{dC_M/d\alpha}{dC_L/d\alpha} \quad (9.15)$$

The aircraft's aerodynamic derivatives $dC_M/d\alpha$ and $dC_L/d\alpha$, evaluated about the CG, are numerically estimated using a finite difference formulation and the methods described in chapter 7. The same method is used even if stabilisers are present, as their effect is factored into the aircraft's aerodynamic derivatives.

The synthesis algorithm ensures that a minimum predefined static margin, defined as a fraction of the reference wing's mean aerodynamic chord, is met by moving the reference wing position. As moving the wing will affect both the neutral point and centre of gravity positions, an iterative approach is used. The minimum static margin is found relative to the aftmost longitudinal CG identified using the methods in section 9.1.2.

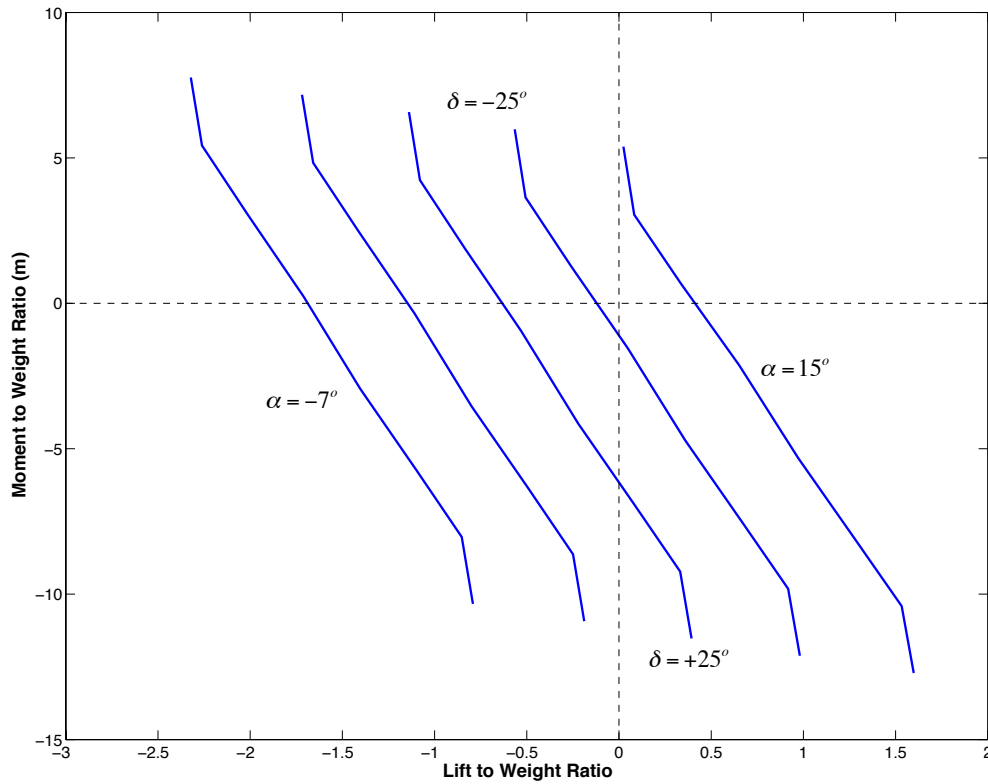


Figure 9.2: Plot of Lift to Weight ratio (L/W) vs. pitching moment to weight ratio (M/W) for varying angles of attack and elevator deflection angles

In flight, the aircraft is considered trimmed when the total pitching moment about the aircraft is equal to zero. The control surface deflection (δ), angle of attack (α) and thrust coefficient (C_T) for

trimmed steady level flight is found by solving the following system of equations.

$$\begin{cases} C_L + \frac{dC_L}{d\delta} \delta + C_T \sin(\alpha + i_p) = \frac{W}{\frac{1}{2} \rho U_\infty^2 S_{ref}} \\ C_{D_o} + KC_L^2 + \frac{dC_D}{d\delta} \delta = \frac{C_T}{\cos(\alpha + i_p)} \\ -\frac{C_L x_{sm}}{\bar{c}} + \frac{C_T}{N_{eng}} \sum_{i=1}^{N_{eng}} \left[\frac{(x_{cg} - x_{eng_i}) \sin i_p + (z_{cg} - z_{eng_i}) \cos i_p}{\bar{c}} \right] + \frac{dC_M}{d\delta} \delta = 0 \end{cases} \quad (9.16)$$

where i_p is the thrust line setting angle to the aircraft baseline.

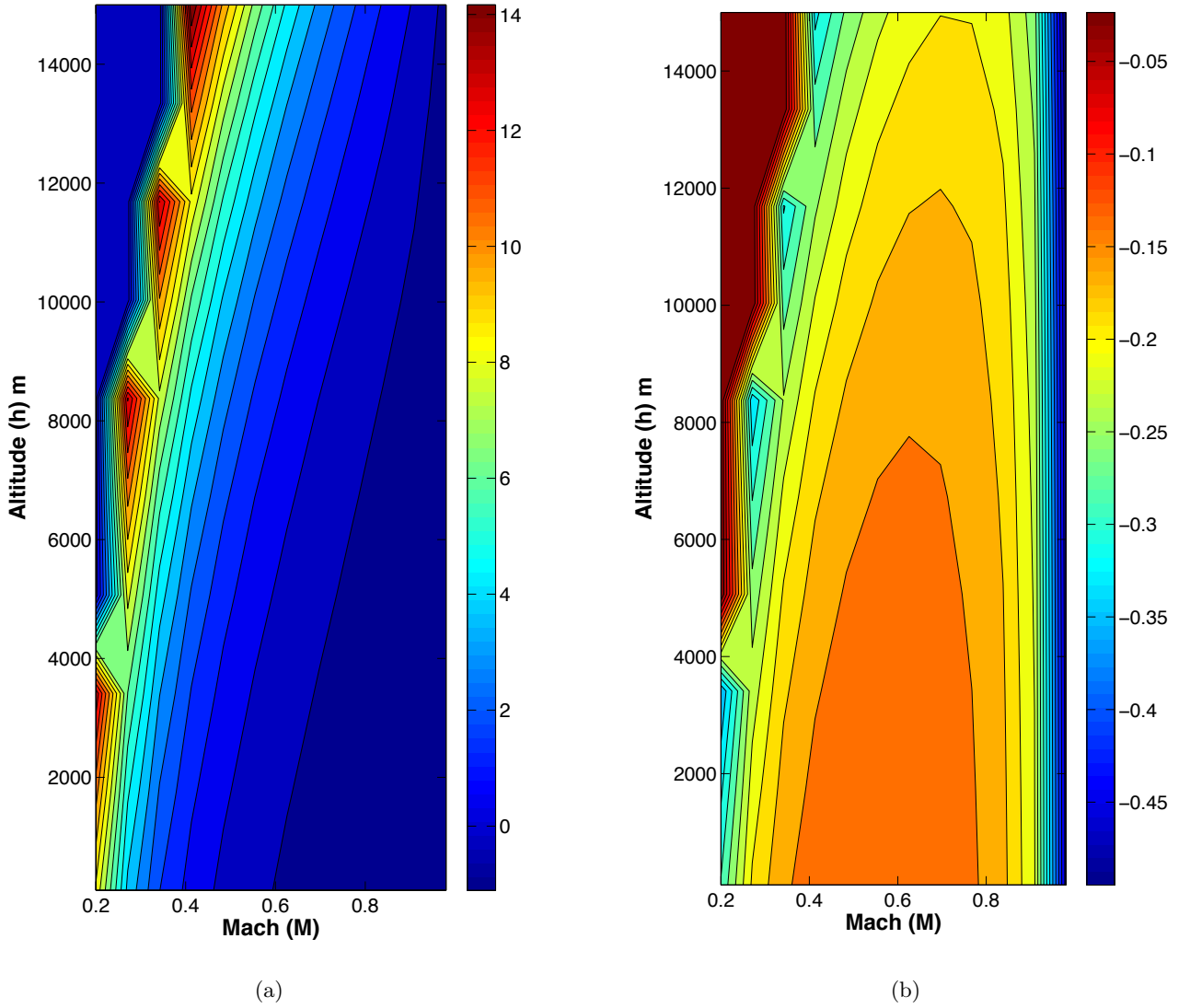


Figure 9.3: Variation of aircraft trimmed (a) angle of attack and (b) percentage elevator deflection for varying cruise altitude and Mach number

A plot of aircraft lift vs. moment, seen in figure 9.2, is typically used to solve eq. (9.16). Also seen in this figure is the nonlinear behaviour of $dC_L/d\delta$ at high control surface deflection angles, as

predicted by the DATCOM methods used. The angle of attack and elevator deflection necessary to trim a sample aircraft at varying altitude and Mach number can be seen in figure 9.3. From this figure it is evident that for an all-wing aircraft, lacking horizontal stabilizers and with even a small positive static margin, the elevators placed alone the trailing edge of the wing must be actuated to negative deflections to trim the aircraft. The associated loss of lift combined with the increased trim drag have a substantial impact on the aircraft's lift to drag ratio. This issue is only exacerbated by the high vertical placement of the aircraft's engines, resulting in an even larger nose down moment to be counteracted.

The issue could be mitigated by allowing a negative static margin to occur during flight. Using modern control systems, the aircraft could be statically unstable and still capable of flight. In such a case the aircraft would be returned to its equilibrium condition by automatically actuating its control surfaces. Such a design choice has obvious performance advantages, however it may be excessively risky for a passenger airliner.

The aircraft must also be laterally statically stable in flight. This condition ensures that the vertical stabilisers used are large enough to ensure that the aircraft's yaw and roll attitudes return to their equilibrium state after a perturbation in yaw is applied. The stability conditions are therefore:

$$\frac{dC_N}{d\beta} > 0 \quad (9.17)$$

$$\frac{dC_L}{d\beta} < 0 \quad (9.18)$$

where N and L are the yaw and roll moments taken about the aircraft's stability axes and β is the sideslip angle. These derivatives are also obtained using a finite difference formulation. If these inequalities are not satisfied, the size of the aircraft's tip fins is increased until a maximum aspect ratio is reached. If the aircraft remains laterally unstable additional fins are placed on the aircraft centre section afterbody.

9.2.2 Hydrostatic

The static stability of a hull at rest in water is determined by its metacentric height. The metacentric height is the vertical distance between the vessel's centre of gravity and its metacentre. The metacentre is the point where the imaginary line through the centre of buoyancy of a hull perturbed in pitch or roll meets the line through the unperturbed centre of buoyancy. For small perturbation

angles, the height between the metacentre (z_M) and the centre of buoyancy (z_B) is given by:

$$z_M - z_B = \frac{I}{V} = \frac{gwI}{W} \quad (9.19)$$

where I is the second moment of area of the waterplane and V is the submerged volume. The centre of buoyancy is the centroid of the submerged volume. The longitudinal metacentre is therefore found using I_{yy} and the lateral one using I_{xx} . Figure 8.3 offers a depiction of a hull's waterplane area.

The metacentric height must be positive in order for the vessel to be statically stable. This condition must hold for both maximum takeoff weight and empty weight conditions. For the seaplane hulls considered, longitudinal static stability has never been an issue. Lateral stability however often had to be enhanced using tip floats. This was often the case due to the high centre of gravity position and low waterplane area width of past seaplane designs. By lowering the centre of gravity through decisions made early in the design process, the resulting aircraft are often stable under all loading conditions.

If a negative metacentric height is found, an occasional occurrence when the vessel's draft is minimised, the metacentric height calculations are repeated for increasing roll angles. The angle at which the metacentric height becomes positive is called the loll angle. A maximum loll angle constraint of 5-10 degrees is set to ensure the aircraft's wingtips are kept clear of the water and that passenger safety and comfort is not affected by excessive roll angles. If the loll angle estimated initially is excessive, the seawing width (y_{sw}), where $y_{le} \geq y_{sw}$, is increased, thus increasing the waterplane width, until the roll constraints are met.

This analysis is applicable as long as hydrostatic pressure forces are dominant. As velocity increases and hydrodynamic forces become important, the hydrodynamic static stability of a prismatic hull is always ensured by the nature of the flow about the hull itself. Longitudinally, a nose up perturbation in pitch would increase the amount of hydrodynamic lift produced, reducing the hull wetted area and therefore moving the centre of pressure closer to the step. The resulting nose down pitching moment would subsequently bring the vessel back to its equilibrium position.

The same holds for lateral stability where a perturbation in roll would result in higher hydrodynamic forces on the side with the lower apparent deadrise angle, producing a righting hydrodynamic rolling moment and returning the hull to its equilibrium roll angle.

9.3 Dynamic Stability

9.3.1 Aerodynamic

Static stability can only describe a fraction of the aircraft's in-flight response to a perturbation. Aircraft often exhibit a consistent set of dynamic modes, the stability of which is determined by the aircraft's aerodynamic characteristics and the equilibrium flight conditions. Typically dynamic stability analysis is performed by numerically finding the aircraft's response to a step input using the aircraft's equations of motion. A simplified approach entails using small disturbance theory to linearise the problem and identify the aircraft's response by finding the eigenvalues and eigenvectors of the resulting stability matrices. In most cases gyroscopic effects are overlooked and the longitudinal and lateral matrices are separated to make analysis of the problem easier.

An extensive analysis of the aircraft's dynamic behaviour is not warranted at this stage in the design process and therefore a set of simple equations, derived from the aforementioned stability matrices using a number of assumptions, are used. These equations are strictly applicable only to conventional aircraft designs, for which basic assumptions used are valid. However as previous studies have shown that the basic assumptions made also hold true for aircraft of design similar to the one under investigation, this simpler approach was taken.

The aircraft's aerodynamic characteristics are defined in terms of stability derivatives (Y_x), which describe the effect a small disturbance in the state x has on the aircraft's Y force or moment when evaluated at the aircraft's equilibrium flight conditions. For conventional aircraft a number of empirical relations or strip theory are used to evaluate the aircraft's aerodynamic derivatives. Using many of these methods for the aircraft in question is impossible, especially for the derivatives where the major contributions typically come from a horizontal or vertical tailplane. Instead the quasi-steady stability derivatives for all the non-accelerating states are evaluated using a finite difference formulation and the methods described in chapter 7.

The VLM code used is capable of modelling varying pitch and sideslip angles as well as any angular rate applied to the aircraft about its CG. The steady nature of the code however prohibits the modelling of accelerating motion and therefore the $Z_{\dot{w}}$ and $M_{\dot{w}}$ stability derivatives. On conventional aircraft these derivatives are often considered to be solely affected by the size of the horizontal tailplane and its distance aft of the main wing, as a measure of the effect that the resulting change in downwash angle has on the tail's lifting characteristics. Therefore as any horizontal tails are mounted on the body, forward of the wing's trailing edge and thus remain relatively unaffected by its wake,

these derivatives are assumed to be zero.

Having found the aircraft's stability derivatives, the dynamic modes can be analysed using the methods given by Roskam [74]. All the stability derivatives quoted are dimensional.

Phugoid The phugoid is a low frequency, lightly damped longitudinal oscillatory mode in pitch and velocity. The undamped natural frequency (ω_n) and damping ratio (ξ) are given by:

$$\omega_n = \frac{1.414g}{aM_\infty} \quad (9.20)$$

$$\xi = -g \frac{2X + aM_\infty X_u}{2\omega_n aM_\infty W} \quad (9.21)$$

where X is the sum of longitudinal forces at the desired flight condition, W is aircraft weight and a is the speed of sound.

SPPO The short period pitch oscillation is a high frequency, highly damped angle of attack, pitch rate oscillation.

$$\omega_n = \sqrt{\frac{gZ_w M_q}{W I_{yy}} - \frac{aM_\infty M_w}{I_{yy}}} \quad (9.22)$$

$$\xi = -\frac{\frac{M_q + M_{\dot{w}}}{I_{yy}} + \frac{gZ_w}{W}}{2\omega_n} \quad (9.23)$$

where q is the rate of pitch about the CG and the w perturbation velocity derivatives are obtained using

$$Z_w = \frac{Z_\alpha}{aM_\infty} \quad (9.24)$$

Dutch Roll The dutch roll is a lateral oscillation where the yaw and roll angles vary out of phase.

$$\omega_n = \sqrt{\frac{gaM_\infty Y_v N_r + N_v(aM_\infty W - gY_r)}{W I_{zz}}} \quad (9.25)$$

$$\xi = -\frac{\frac{N_r}{I_{zz}} + \frac{gY_v}{W}}{2\omega_n} \quad (9.26)$$

where r is the yaw rate, N is the yaw moment and Y the side force in the spanwise direction. Assuming small angles, the derivatives with respect to a spanwise velocity perturbation v can be found in terms of the sideslip angle (β) as:

$$Y_v = \frac{Y_\beta}{aM_\infty} \quad (9.27)$$

Spiral The spiral mode is a non-oscillatory instability that causes the aircraft to go into a spiral dive following a perturbation in roll angle. The stability constraint is:

$$L_v N_r - N_v L_r > 0 \quad (9.28)$$

The time to double or half amplitude, depending on whether eq. (9.28) is satisfied, is given by:

$$T = \left| \frac{\ln 2 \ L_v \ I_{zz}}{N_v L_r - L_v N_r} \right| \quad (9.29)$$

For the oscillatory modes, the time to half or double amplitude can be found using the undamped natural frequency and damping ratio as:

$$T = \left| \frac{\ln 2}{\xi \ \omega_n} \right| \quad (9.30)$$

9.3.2 Hydrodynamic

During takeoff and landing seaplane hulls are susceptible to a longitudinal dynamic instability commonly known as porpoising, a pitch-heave instability triggered when the hull's running trim exceeds a critical value. Figure 9.4 shows the variation of trim with velocity for varying elevator deflection angles and the critical porpoising trim angles for a 1:52.7 scale model of a proposed large transonic jet seaplane. The unstable oscillatory motion triggered when planing outside the stable trim boundaries is not only extremely uncomfortable for passengers but can result in very high impact loads which can damage the hull.

Being able to identify these trim limits for an arbitrary hull shape is therefore imperative so that the vessel's running trim and trim limits can be adjusted if necessary by moving the centre of gravity with respect to the step, moving the wing or increasing the elevator control's authority at low speeds.

To remain stable the hull's equilibrium trim, found using the methods in section 8.3.3, must lie between the minimum and maximum trim limits. The minimum trim limit starts near the hump speed ($C_v \approx \sqrt{(C_\Delta)/0.3}$) and keeps decreasing with increasing velocity until liftoff. At speeds higher than the hump, it represents the onset of a porpoising mode driven solely by oscillating pressure loads acting on the forebody in the region of the main step, as the afterbody is not wetted at such low trim angles. This instability has been extensively studied as it is also encountered by speedboats and numerical estimates of the critical trim angles using the added mass method presented by Martin [51] have shown some promise. At lower velocities however the afterbody is not fully clear of the water

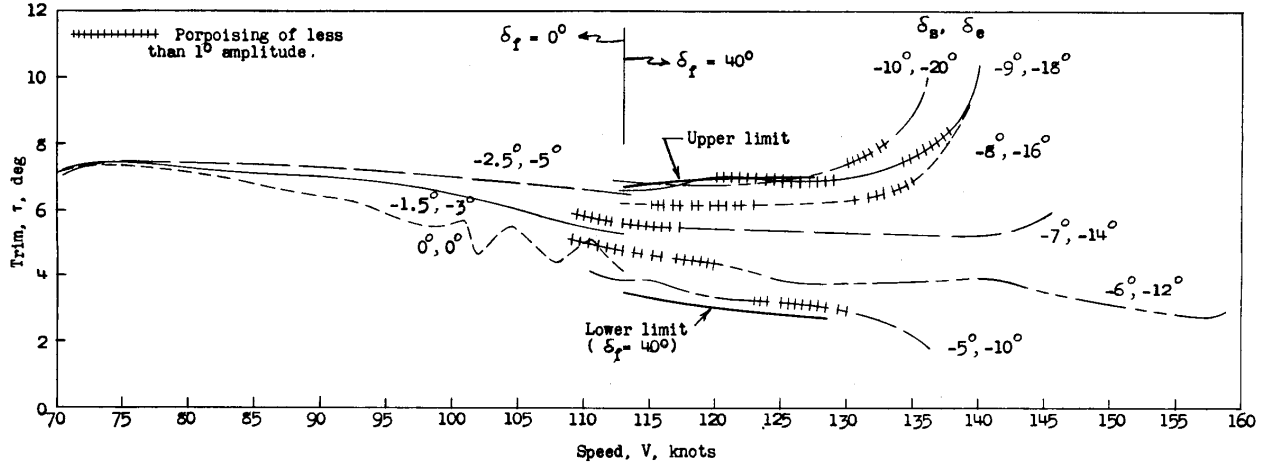


Figure 9.4: Porpoising stability limits and running trim angles for varying elevator and flap deflections [42]

and can affect the lower porpoising trim limit, making a theoretical analysis of porpoising at those speeds impossible.

The upper trim stability limit represents the onset of a joint forebody-afterbody instability, triggered by the wetting of the afterbody at high angles of attack. It is mostly affected by the geometry of the main step, the length of the afterbody and its keel angle. At high speeds the upper stability limit may also represent the onset of skipping, whereby high instantaneous loads may propel the aircraft into the air but lacking sufficient levels of aerodynamic lift it plunges back into the water. The onset of the forebody-afterbody instability mode could be predicted using the added mass methods referred to previously, however no accurate ways of modelling the hull's wake aft of a step, let alone its dynamic behaviour, could be found.

Since the 1940's both NACA and the Aeronautical Research Council (ARC) have published numerous reports detailing the experimental results of tank tests on dynamically similar flying boat scale models. The tests were conducted either by testing the model at constant speed and different loading and trim conditions or by simulating the hull's takeoff and landing in accelerated runs.

Early American and all available UK constant speed studies perturbed the model's running trim by 2 degrees and identified the regions where the model's motion became unstable. Using this method a single upper and lower trim stability limit was identified. Later NACA studies however identified the trim limits by progressively increasing the hull's trim using the model's elevator controls. The resulting lower limit was identical to that found using the perturbation method. For the upper trim

limit however, two distinct trim levels were identified. The increasing trim limit identifies the hull trim at which instability is first observed when the elevator deflection is increased and is considered a reasonable estimate of the hull's upper stability trim limit during takeoff. As soon as the hull becomes unstable, the elevator angle is decreased until the hull becomes stable again, at the decreasing trim stability limit. The decreasing trim stability limit is typically considered to represent the upper stability limit encountered during landings. Further information on the experimental methods used and their accuracy is given by Smith and Allen [85] and Olson and Land [63].

As a theoretical approach was unavailable, experimental data were taken from the reports detailed in appendix F in order for an empirical method to be produced. Similar equations have already been produced for planing hulls however they are not applicable to seaplane design as they neglect the hydrodynamic effects of the afterbody and the existence of important aerodynamic stability derivatives.

The effects of the various parameters given in section 8.3.1 can be seen from the results of a systematic set of experiments performed on a dynamically similar 1:30 scale model of the XPB2M-1 flying boat [21]. Both the upper and lower stability limits were found to be proportional to the seaplane's weight. The centre of gravity position although a major factor in determining the hull's equilibrium trim attitude was found to have no effect on the onset of porpoising. The hull's moment of inertia was found to have a minor effect on the lower trim limit at the hump speed, however given the relative insignificance and the lack of information on each experimental model's inertial properties, this parameter was omitted.

The hull shape has a major effect on the critical trim angles. Reducing the afterbody keel angle, increasing the afterbody length or increasing the hull's main step height to beam ratio leads to a reduction in both the upper trim limit and the maximum value of the lower limit. The hull's forebody length and deadrise angles have less predictable effects. Forebody warping has also been found to affect the lower trim limit. Given the different deadrise angle distributions used in the models examined, an average forebody warp value was used:

$$\frac{d\beta}{dB} = B \left(\frac{\beta_{10\%} - \beta_1}{0.9L_f - L_{flat}} \right) \quad (9.31)$$

where $\beta_{10\%}$ is the hull's deadrise angle at $x/B = 10\%L_f/B$ (including the effects of chine flare).

The seaplane's aerodynamic surfaces also affect the porpoising trim limits. Increasing the value of the aircraft's M_q damping derivative mostly affects the lower trim limit, increasingly lowering it as

velocity increases. The lower trim limit at the hump however remains unaffected. For an aircraft with a horizontal stabiliser, the non-dimensional pitching moment derivative with respect to rate of pitch is given by:

$$M'_q = \frac{-2M_q}{wU_\infty B^4} \approx \frac{\rho l_T^2 S_T (C_{L_\alpha})_T}{wB^4} \quad (9.32)$$

where l_T is the distance between the aircraft's centre of gravity and the horizontal tailplane's aerodynamic centre and S_T is the tailplane reference area. Also critical is the aircraft's Z_θ derivative:

$$Z'_\theta = \frac{2Z_\theta}{wU_\infty^2 B^2} = \frac{\rho S_{ref} C_{L_\alpha}}{wB^2} \quad (9.33)$$

For a seaplane which lacks horizontal tailplanes, the stability derivatives are found using a finite difference method in conjunction with the numerical methods described in chapter 7. If the aerodynamic characteristics of the test models used for individual tank tests were not supplied, the aerodynamic derivatives were found using eqs. (7.1), (9.32) and (9.33).

The effect that the step planform has on the porpoising limits is unknown, consequently the step angle was removed from the step height parameter. Therefore the remaining hull shape parameters for $N_{st} = 1 - 2$ used are:

$$\begin{aligned} L'_f &= L_f/B \\ h'_1 &= h_1/B \\ dL'_1 &= \left[(N_{st} - 1) \left(\frac{L_2}{B} - \frac{L_f}{B} \right) - (N_{st} - 2) \left(\frac{L}{B} - \frac{L_f}{B} \right) \right]^{(1.5N_{st} - 0.5N_{st}^2)} \\ \cos \beta'_2 &= (\cos \beta_2)^{(0.5N_{st}^2 - 0.5N_{st})} \\ \tan \delta'_{k_2} &= (\tan \delta_{k_2})^{(0.5N_{st}^2 - 0.5N_{st})} \\ h'_2 &= (h_2/B)^{(0.5N_{st}^2 - 0.5N_{st})} \\ \sin \delta'_{s_2} &= (\sin \delta_{s_2})^{(0.5N_{st}^2 - 0.5N_{st})} \\ dL'_2 &= \left(\frac{L}{B} - \frac{L_2}{B} \right)^{(0.5N_{st}^2 - 0.5N_{st})} \\ \cos \beta'_3 &= \cos \beta_{ab} \end{aligned}$$

The strict applicability range for the empirical methods reduced can be seen from the graphs in appendix G. The accuracy of results obtained using eqs. (9.34) to (9.36) in predicting the stability limits of a full scale aircraft is unknown. Comparing experimental results, which the empirical methods derived are based on, to data from full scale aircraft tests, Smith and White [86] found that the trim

limits predicted using scale models were generally higher than those of the real aircraft. On average the lower trim limit was over predicted by 0-3 degrees and the upper trim limit by 0.5-4.5 degrees. The differences were partly attributed to the disturbance method used and results obtained using the later NACA elevator based method were deemed more accurate, providing the seaplane operates in conditions where no large swells are present.

9.3.2.1 Lower Trim Limit

A total of 2892 data points were collected for the lower trim limit. Using the stepwise regression approach detailed previously and a total of 22 parameters, a fit with $R^2 = 0.876$ and an average absolute percent error of approximately 16% was produced. This level of error however is acceptable due to the low average trim value of the input data.

$$\begin{aligned}
\tau_{min} = & a_o + a_1 \frac{Z_\theta'^2 M_q' \cos \beta_1 \cos \beta_3'}{L_f' dL_2' \sin \delta_{s1} \tan \delta_{k2}'} + a_2 \frac{Z_\theta' \cos \beta_3'}{L_f' dL_2'} + a_3 \frac{\cos \beta_1 Z_\theta'}{L_f' \sqrt{C_\Delta} C_v \sin \delta_{s1} \cos \beta_3'} + \\
& + a_4 \frac{\sqrt{C_\Delta} \cos \beta_3'}{C_v \cos \beta_1} + a_5 \frac{\sqrt{C_\Delta} \cos \beta_1}{C_v^3 L_f' \sin \delta_{s1} \cos \beta_3'} + a_6 \frac{Z_\theta' \sin \delta_{s1}}{L_f' \cos \beta_1 \cos \beta_2'} + \\
& + a_7 \frac{M_q' \sin \delta_{s2}'}{\sqrt{C_\Delta} C_v L_f' dL_1' \cos \beta_1'} + a_8 \frac{M_q' \cos \beta_1' \sin \delta_{s2}'}{\sqrt{C_\Delta} C_v L_f' dL_1' \cos \beta_2'} + a_9 \frac{C_\Delta^{3/2} dL_1' \sec \beta_1}{C_v^5 L_f' \cos \beta_2' \sin \delta_{s2}'} + \\
& + a_{10} \frac{C_\Delta^{3/2} dL_1' \sec \beta_1 \sec \beta_2'}{C_v^5 L_f' \tan \delta_{k1} \sin \delta_{s2}'} \left(\frac{d\beta}{dB} \right) + a_{11} \frac{dL_1' \cos \beta_2' \cos \beta_3'}{L_f' \tan \delta_{k1}} + \\
& + a_{12} \frac{\cos \beta_1 dL_1' \cos \beta_3' Z_\theta'}{L_f' dL_2' \tan \delta_{k1} \cos \beta_2' \tan \delta_{k2}'} + a_{13} \frac{C_\Delta L_f' \cos \beta_2' \cos \beta_3'}{dL_1' \tan \delta_{k1} \cos \beta_1 dL_2' \tan \delta_{k2}'} + \\
& + a_{14} \frac{C_\Delta Z_\theta' \cos \beta_2'}{dL_1' dL_2' \tan \delta_{k2}' \cos \beta_3'} + a_{15} \frac{dL_2' \tan \delta_{k2}' \cos \beta_3' M_q'^2}{\sqrt{C_\Delta} C_v L_f' dL_1' \cos \beta_1 \cos \beta_2'} \left(\frac{d\beta}{dB} \right)^2 + \\
& + a_{16} \frac{\cos \beta_2'}{\sqrt{C_\Delta} C_v L_f' dL_1' \cos \beta_1 \tan \delta_{k2}'} \left(\frac{d\beta}{dB} \right)^2 + a_{17} \frac{dL_1' dL_2' \cos \beta_1 \cos \beta_2' \cos \beta_3' \tan \delta_{k2}'}{\sqrt{C_\Delta} C_v L_f' \tan \delta_{k1}} + \\
& + a_{18} \frac{dL_1' dL_2' \cos \beta_2' \tan \delta_{k2}'}{\sqrt{C_\Delta} C_v \cos \beta_1 \cos \beta_3'} \left(\frac{d\beta}{dB} \right) + a_{19} \frac{C_\Delta^{3/2} L_f' \cos \beta_1}{C_v dL_1' dL_2' \tan \delta_{k1} \tan \delta_{k2}' \cos \beta_3'} + \\
& + a_{20} \frac{C_\Delta^{3/2} \cos \beta_3'}{C_v^5 L_f' \cos \beta_1 \cos \beta_2'} + a_{21} \frac{C_\Delta^{3/2} dL_1' \cos \beta_2' \cos \beta_3' Z_\theta'}{C_v^5 L_f' \cos \beta_1} \left(\frac{d\beta}{dB} \right) + \\
& + a_{22} \frac{C_\Delta^{3/2} \cos^2 \beta_1 \sin^2 \delta_{s1}}{C_v^3} \left(\frac{d\beta}{dB} \right)^2
\end{aligned} \tag{9.34}$$

a_0	2.953724	a_6	24.288982	a_{12}	0.962785	a_{18}	0.105365
a_1	-36.820960	a_7	839.55079	a_{13}	-0.097771	a_{19}	0.569929
a_2	-145.999150	a_8	-1112.2843	a_{14}	-42.761578	a_{20}	-3946.3485
a_3	92.964938	a_9	247.869130	a_{15}	1.258753	a_{21}	-1078.1504
a_4	34.369388	a_{10}	-6.050226	a_{16}	-0.021530	a_{22}	11.545650
a_5	525.306150	a_{11}	-0.313854	a_{17}	-0.592827		

Table 9.1: Coefficients necessary to evaluate eq. (9.34) to obtain lower porpoising trim stability limit

9.3.2.2 Upper Trim Limit - Increasing Trim

The increasing trim upper trim limit relation (9.35) was generated based on 1955 experimental observations. The upper trim stability limits obtained by perturbing the model by 2 degrees in pitch, although obtained using a different experimental methodology, were found to agree well with the increasing trim, upper limits for similar aircraft and were thus incorporated into the dataset used. The resulting model uses 25 parameters to produce a fit with $R^2 = 0.8071$ and a mean absolute percent error of 6.40%.

$$\begin{aligned}
\tau_{max-inc} = & a_0 + \left(\frac{d\beta}{dB}\right)^2 \left[\frac{Z'_\theta}{h'_1 h'_2 \sin \delta_{s1}} \left(a_1 + \frac{a_2}{\cos \beta_1} \right) + a_3 \frac{h'_1 M_q'^2 \cos \beta_1 \cos \beta'_3}{C_\Delta dL'_2 \sin \delta_{s1} \tan \delta'_{k2}} \right] + \\
& + a_4 \frac{Z_\theta'^2 \tan \delta'_{k2}}{L'_f h'_1 \cos \beta_1 \cos \beta'_3} + a_5 \frac{h'_1 \sin \delta_{s1} \cos \beta_1 \cos \beta'_3}{L'_f dL'_2} + a_6 \frac{h'_1 dL'_2 \tan \delta'_{k2} \cos \beta'_3}{\sqrt{C_\Delta} C_v \sin \delta_{s1}} + \\
& + a_7 \frac{h'_1 \cos \beta_1 \sin \delta_{s1} dL'_2 \tan \delta'_{k2}}{\sqrt{C_\Delta} C_v L'_f \cos \beta'_3} + \left(\frac{d\beta}{dB}\right)^2 \left[a_8 \frac{C_\Delta \tan \delta_{k1} Z_\theta'^2}{L'_f dL'_1 \cos \beta_1 \cos \beta'_2 \sin \delta'_{s2}} + \right. \\
& + a_9 \frac{C_\Delta}{dL'_1 \cos \beta_1 \cos \beta'_2 \sin \delta'_{s2}} + a_{10} \frac{C_\Delta L'_f \cos \beta_1 \tan \delta_{k1} \cos \beta'_2}{\sin \delta'_{s2}} \left. \right] + \\
& + a_{11} \frac{C_\Delta \tan \delta_{k1}}{L'_f dL'_1 \cos \beta_1 \cos \beta'_2 \sin \delta'_{s2}} + a_{12} \frac{C_\Delta \sin \delta'_{s2} M_q'}{dL'_1 h'_2 \cos \beta_1 \cos \beta'_2} + \\
& + \left(\frac{d\beta}{dB}\right)^2 \left[a_{13} \frac{L'_f h'_2 \sin \delta'_{s2}}{\sqrt{C_\Delta} C_v \cos \beta'_2} + a_{14} \frac{C_\Delta^{3/2} \cos \beta_1 M_q'^2}{C_v h'_2 L'_f dL'_1 \tan \delta_{k1} \cos \beta'_2} + \right. \\
& + a_{15} \frac{\sqrt{C_\Delta} Z'_\theta dL'_1 \cos \beta_1}{C_v^3 \cos \beta'_2 \sin \delta'_{s2}} \left. \right] + a_{16} \frac{\sqrt{C_\Delta} L'_f \cos \beta_1 \tan \delta_{k1}}{C_v^3 dL'_1 \cos \beta'_2} + \\
& + a_{17} \frac{C_\Delta^{3/2} (d\beta/dB) \sec \beta'_3}{C_v^3 L'_f \cos \beta_1 dL'_1 \tan \delta_{k1} \sin \delta'_{s2}} + a_{18} \frac{\cos \beta_1 \cos \beta'_2 \cos \beta'_3}{dL'_1 dL'_2 \tan \delta_{k1}} + \\
& + a_{19} \frac{\cos \beta_1 dL'_2 \tan \delta'_{k2}}{\sqrt{C_\Delta} C_v \tan \delta_{k1} \cos \beta'_3} + a_{20} \frac{\sqrt{C_\Delta} \tan \delta_{k1} \sec \beta'_3}{dL'_1 \cos \beta'_1 \cos \beta'_2} + a_{21} \frac{\sqrt{C_\Delta} \tan \delta_{k1}}{C_v dL'_1 \cos \beta'_2 \cos \beta'_3} + \\
& + a_{22} \frac{\sqrt{C_\Delta} dL'_2 \tan \delta_{k1} \cos \beta'_3}{C_v^3 L'_f dL'_1 \cos \beta'_1 \cos \beta'_2} + \left(\frac{d\beta}{dB}\right)^2 \left[a_{23} h'_2 L'_f \tan \delta_{k1} \cos \beta'_2 + \right. \\
& + a_{24} Z'_\theta L'_f \cos^2 \beta_1 dL'_2{}^2 \tan^2 \delta_{k1} \tan \delta'_{k2} \cos^2 \beta'_3 \left. \right] + a_{25} \cos^2 \beta_1 \cos \beta'_2 \cos \beta'_3
\end{aligned} \tag{9.35}$$

a_0	14.513148	a_6	71.572430	a_{12}	-4.451600	a_{18}	-1.559288
a_1	-0.002954	a_7	-236.751280	a_{13}	0.015349	a_{19}	-0.361557
a_2	0.003404	a_8	-15.818928	a_{14}	0.108767	a_{20}	-1642.5325
a_3	-0.001711	a_9	-0.0080715	a_{15}	0.572640	a_{21}	2913.3896
a_4	-21.681416	a_{10}	0.006341	a_{16}	-1721.0386	a_{22}	-1852.2205
a_5	96.721479	a_{11}	-61.870418	a_{17}	-0.438993	a_{23}	-0.032396
		a_{24}	0.407889	a_{25}	-7.305247		

Table 9.2: Coefficients necessary to evaluate eq. (9.35) to obtain the increasing trim, upper porpoising stability trim limit

9.3.2.3 Upper Trim Limit - Decreasing Trim

The number of data points available to build a decreasing trim upper trim limit relation (9.35) was substantially less than those used in other case. Using a total of 637 points, including data from tank tests simulating landing conditions, an expression using 25 parameters was produced, the fit having an $R^2 = 0.8067$ and a mean absolute percent error of 7.62%. However care should be taken in using this relation as the low number of data points and the large number of predictors may have led to overfilling and loss of generality of the model.

a_0	11.959269	a_6	-0.034042	a_{12}	0.005893	a_{18}	499.967570
a_1	-0.057480	a_7	-29.303595	a_{13}	602.892310	a_{19}	-31.784055
a_2	43.603971	a_8	2209.6365	a_{14}	-9.798781	a_{20}	-2054.5784
a_3	0.181960	a_9	37.511237	a_{15}	-0.099445	a_{21}	-570.445600
a_4	224.554340	a_{10}	-0.026031	a_{16}	-0.983849	a_{22}	0.006228
a_5	0.207037	a_{11}	-0.291656	a_{17}	0.130966	a_{23}	-0.237511
		a_{24}	0.460156	a_{25}	24.959765		

Table 9.3: Coefficients necessary to evaluate eq. (9.36) to obtain the decreasing trim, upper porpoising stability trim limit

$$\begin{aligned}
\tau_{max-dec} = & a_0 + a_1 \frac{L'_f \cos \beta_1}{h'_1 \cos \beta'_3} + \left(\frac{d\beta}{dB} \right) \left[a_2 \frac{h'_1 \sin \delta_{s_1} Z'_\theta}{L'_f dL'_2 \cos \beta_1 \cos \beta'_3} + a_3 \frac{Z'_\theta L'_f \tan \delta'_{k_2} \cos \beta'_3}{\sqrt{C_\Delta} C_v h'_1 \sin \delta_{s_1} \cos \beta_1} + \right. \\
& + a_4 \frac{Z'_\theta h'_1 \tan \delta_{k_2} \cos \beta_1}{\sqrt{C_\Delta} C_v L'_f \cos \beta'_3} + a_6 \frac{C_\Delta^{3/2} dL'_2 \cos \beta_1}{C_v h'_1 \tan \delta'_{s_2}} + a_8 \frac{C_\Delta^{3/2} Z'_\theta \cos \beta'_3}{C_v^5 h'_1 L'_f \cos \beta_1} + \\
& + a_{11} \frac{C_\Delta Z'_\theta \sin \delta'_{s_2}}{h'_2 dL'_1 \tan \delta_{k_1} \cos \beta_1 \cos \beta'_2} + a_{16} \frac{dL'_1 dL'_2 \tan \delta'_{k_2} \cos \beta'_3 Z'_\theta}{\sqrt{C_\Delta} C_v L'_f \tan \delta_{k_1}} + \\
& + a_{19} \frac{\sqrt{C_\Delta} L'_f dL'_2 \tan \delta'_{k_2}}{C_v^3 dL'_1 \cos \beta'_2} \left. \right] + \left(\frac{d\beta}{dB} \right)^2 \left[a_5 \frac{C_\Delta^{3/2} \sin \delta_{s_1} \cos \beta'_3 Z'_\theta}{C_v h'_1 L'_f \cos \beta_1 \tan \delta'_{k_2}} + \right. \\
& + a_7 \frac{C_\Delta^{3/2} Z'_\theta \cos \beta_1 \sin \delta_{s_1}}{C_v^3 h'_1 L'_f dL'_2 \cos \beta'_3} + a_9 \frac{C_\Delta^{5/2} \cos \beta_1 \cos \beta'_3}{C_v^5 \sin \delta_{s_1}} + a_{10} \frac{C_\Delta \sin \delta'_{s_2} \sec \beta'_2}{h'_2 L'_f dL'_1 \tan \delta_{k_1} \cos \beta_1} + \\
& + a_{12} \frac{\sqrt{C_\Delta} L'_f \sin \delta'_{s_2} \cos \beta_1}{C_v h'_2 dL'_1 \tan \delta_{k_1} \cos \beta'_2} + a_{15} \frac{C_\Delta dL'_2 M'_q \cos \beta'_3}{L'_f dL'_1 \tan \delta_{k_2} \cos \beta_1 \cos \beta'_2} + \\
& + a_{17} \frac{\sqrt{C_\Delta} L'_f dL'_2 \tan \delta'_{k_2} \cos \beta_1}{C_v dL'_1 \cos \beta'_2 \cos \beta'_3} + a_{20} \frac{C_\Delta^{3/2} \tan \delta_{k_1} \cos \beta_1 Z'^2_\theta}{C_v^5 L'_f dL'_2 \tan \delta'_{k_2} \cos \beta'_2 \cos \beta'_3} + \\
& + a_{22} \frac{L'_f \sin^2 \delta_{s_1} \tan \delta'_{k_2}}{\sec^2 \beta'_3} + a_{23} \frac{L'_f \tan \delta_{k_1} \tan \delta'_{k_2}}{(\sec \beta_1 \sec \beta'_2 \sec \beta'_3)^2} + a_{24} \frac{dL'_1 dL'^2_2 \tan^2 \delta_{k_1} \tan^2 \delta'_{k_2}}{\sec \beta_1 \sec \beta'_2 \sec^2 \beta'_3} + \\
& + a_{25} \frac{\sqrt{C_\Delta} (dL'_2 \tan \delta_{k_1} \tan \delta'_{k_2})^2}{C_v \sec \beta'_3} \left. \right] + a_{13} \frac{C_\Delta^{5/2} L'_f \sin \delta_{s_2}}{C_v^5 h'_2 \cos \beta_1 \cos \beta'_2} + a_{14} \frac{Z'_\theta \cos \beta'_2 \cos \beta'_3}{L'_f \tan \delta_{k_1} \cos \beta_1} + \\
& + a_{18} \frac{\sqrt{C_\Delta} M'_q dL'_2 \cos \beta_1 \cos \beta'_3}{C_v^3 L'_f dL'_1 \tan \delta_{k_1} \tan \delta'_{k_2}} + a_{21} \frac{C_\Delta^{3/2} dL'_2 \cos \beta_1}{C_v^5 dL'_1 \tan \delta_{k_1} \tan \delta'_{k_2} \cos \beta'_2 \cos \beta'_3}
\end{aligned} \tag{9.36}$$

Chapter 10

Performance

10.1 Engine Performance

To evaluate the flight, takeoff and landing performance of the aircraft in question, the performance of its jet engines must be known. Engine manufacturers often provide graphs relating the maximum uninstalled thrust that can be produced at varying altitudes and Mach numbers. However considering the flexibility of the synthesis method used, the choice to design an engine specifically for each case was made. The high bypass ratio engine design methodology given by Mattingly et al. [52] was chosen, as it has shown sufficient accuracy in predicting the uninstalled thrust output and specific fuel consumption of a parametric engine. Furthermore using this method allows the user to predict the performance of an arbitrary state of the art engine, rather than extrapolating the engine performance from existing ones.

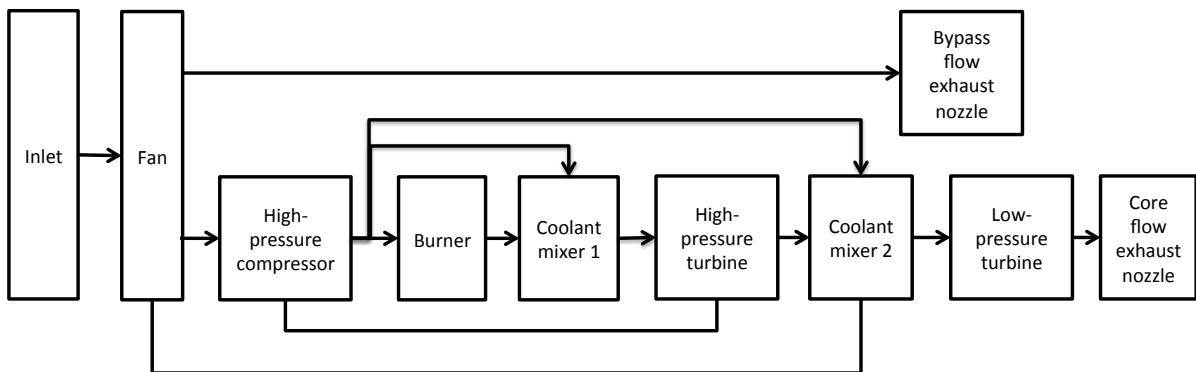


Figure 10.1: Engine architecture assumed for engine design and performance estimation

The engine design methodology used is based on a twin spool turbofan design, shown in figure

10.1. First the engine itself is designed at the chosen design point using an implementation of the ONX method. The algorithm returns the engine's on design thrust per unit mass flow rate ($F/\dot{m}_o$) by propagating the flow through the individual components. It does so for a predefined set of fan and compressor pressure ratios, bypass ratio, bleed air fractions and component figures of merit. The flow's total temperature when exiting the combustion chamber is used to represent the engine's throttle setting.

The engine design mass flow rate is subsequently found, allowing the fan diameter to be calculated using the methods presented in section 6.3. The off-design performance analysis algorithm, an implementation of OFFX, uses the design values previously calculated to iteratively determine the thrust, mass flow rate and specific fuel consumption of the engine at alternate altitudes, free stream Mach numbers and throttle settings. The equations and algorithm flowcharts used for the on and off-design engine analyses can be found in reference [52].

The specific fuel consumption values, given by the on and off-design engine performance evaluation methods, when using the suggested component efficiency values, have been found to be overestimated compared to those of current state of the art engines as there has been a substantial improvement in the various engine component thermal efficiencies. A constant correction factor, based on the expected maximum thrust sfc was therefore used.

The thrust value given by the off-design engine analysis method is the engine's uninstalled thrust. To find the engines installed thrust characteristics, the thrust value found must be corrected for inlet pressure losses. The method given by Roskam [72] was used, not considering bleed air and power extraction corrections which are already accounted for in the engine analysis algorithm. For a straight through circular inlet and podded engines, the installed thrust (T_{inst}) is given by:

$$T_{inst} = T_{uinst} \left(1 - 0.35 K_t M_\infty \frac{\Delta p_{inl}}{q_\infty} \right) \quad (10.1)$$

where

$$\frac{\Delta p_{inl}}{q_\infty} = \frac{f C_f I}{\mu_{inl}^2} \quad (10.2)$$

The duct integral I , estimated along the inlet duct length is given by:

$$I = \int_0^{L_{inl}} \left(\frac{A_{inl}}{A} \right)^2 \frac{\pi D}{A} dL \quad (10.3)$$

where A_c is the inlet area, A and D are the duct's area and diameter at a given length L and μ_{inl} is given by:

$$\mu_{inl} = \frac{A_{inl}}{A_\infty} \quad (10.4)$$

,where A_∞ is the stream tube freestream capture area, calculated using the compressible isentropic continuity relations. The skin friction coefficient C_f is the equivalent flat plate value for a length scale $L = L_{inl}$. The factors f and K_t are found interpolating from digitised graphical values taken from Roskam [72].

Thrust losses caused by the extraction of bleed air and other mechanical components, are already considered by the ONX and OFFX algorithms and are therefore not corrected for again.

Once the initial sizing process is completed, the engines are designed. The engine is initially sized to operate at 80% throttle halfway through the main cruise segment. Off design analyses are subsequently performed to ensure that there is sufficient excess thrust for the takeoff manoeuvre, all other cruise and loiter conditions and that the maximum Mach number can be reached at the desired altitude. If any of these conditions are not met, the cruise throttle setting is gradually decreased until a minimum of 60% is reached. If the excess thrust requirements are still not exceeded, the design point is changed to the condition requiring the maximum thrust and the process is repeated. Alternatively if excessive thrust is available for all the test conditions, the design case throttle setting is increased. A flowchart of the process followed to design the engine can be seen in figure 10.3.

The outputs from the off-design analysis have been collected in figure 10.2 to show the predicted variation of maximum installed and uninstalled total thrust that an aircraft with 3 high bypass ratio turbofan engines produces for varying cruise Mach numbers and altitudes.

The ONX and OFFX implementations were validated against sample outputs in Raymer [67].

10.2 Specific Excess Power

A simple yet efficient way of determining the aircraft's entire flight envelope is the calculation of the aircraft's specific excess power (P_s). The specific excess power is defined as:

$$P_s = aM_\infty \left(\frac{T - D}{W} \right) \quad (10.5)$$

where the drag (D) is found using the trim analysis method described in section 9.2.1 and T is the maximum installed thrust produced at the desired flight conditions.

Looking at an aircraft's P_s curves, such as the one in figure 10.4, can identify the aircraft's ceiling, maximum speed limit and stall speed at varying altitudes. The contour plot can also be used to estimate the aircraft's minimum time to climb, as the specific excess power is equal to an aircraft's maximum climb rate at the given flight conditions.

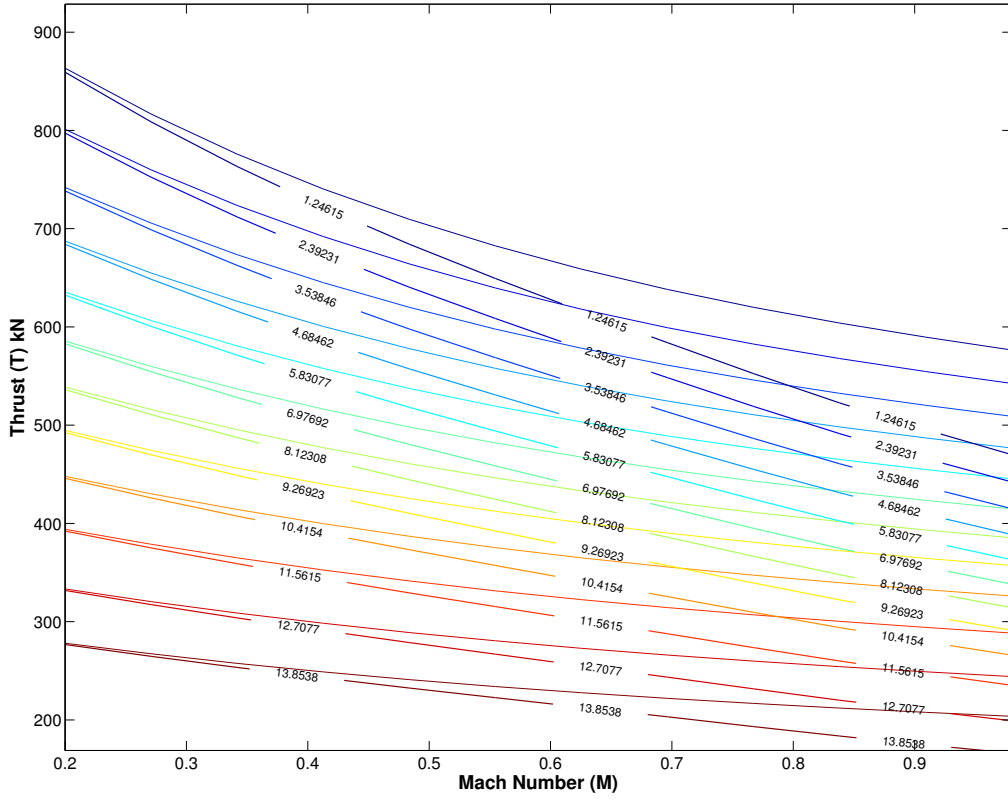


Figure 10.2: Plot of total installed (labeled) and uninstalled thrust vs. mach number for varying altitudes (km)

10.3 Mission Analysis

At the end of the design process, the aircraft's ability to meet the input mission profile requirements must be determined. To abide by the relevant regulations, the fuel necessary to complete the predefined diversion leg of the mission profile is found. The reserve fuel allowance, typically about 5% of the total fuel load, is also withheld from any operational use. The weight fractions (WF) of the remaining cruise and loiter segments are subsequently recalculated to account for the fuel necessary to complete the diversion leg, while the weight fractions for all other segments are kept constant. Any percentage change is applied equally to all the affected mission segments.

The optimum Mach number for the cruise and loiter segments is calculated by using the minimum between the predefined cruise Mach number and the maximum Mach number which can be reached with a predefined throttle setting (typically about 80% for cruise). To estimate the range of the

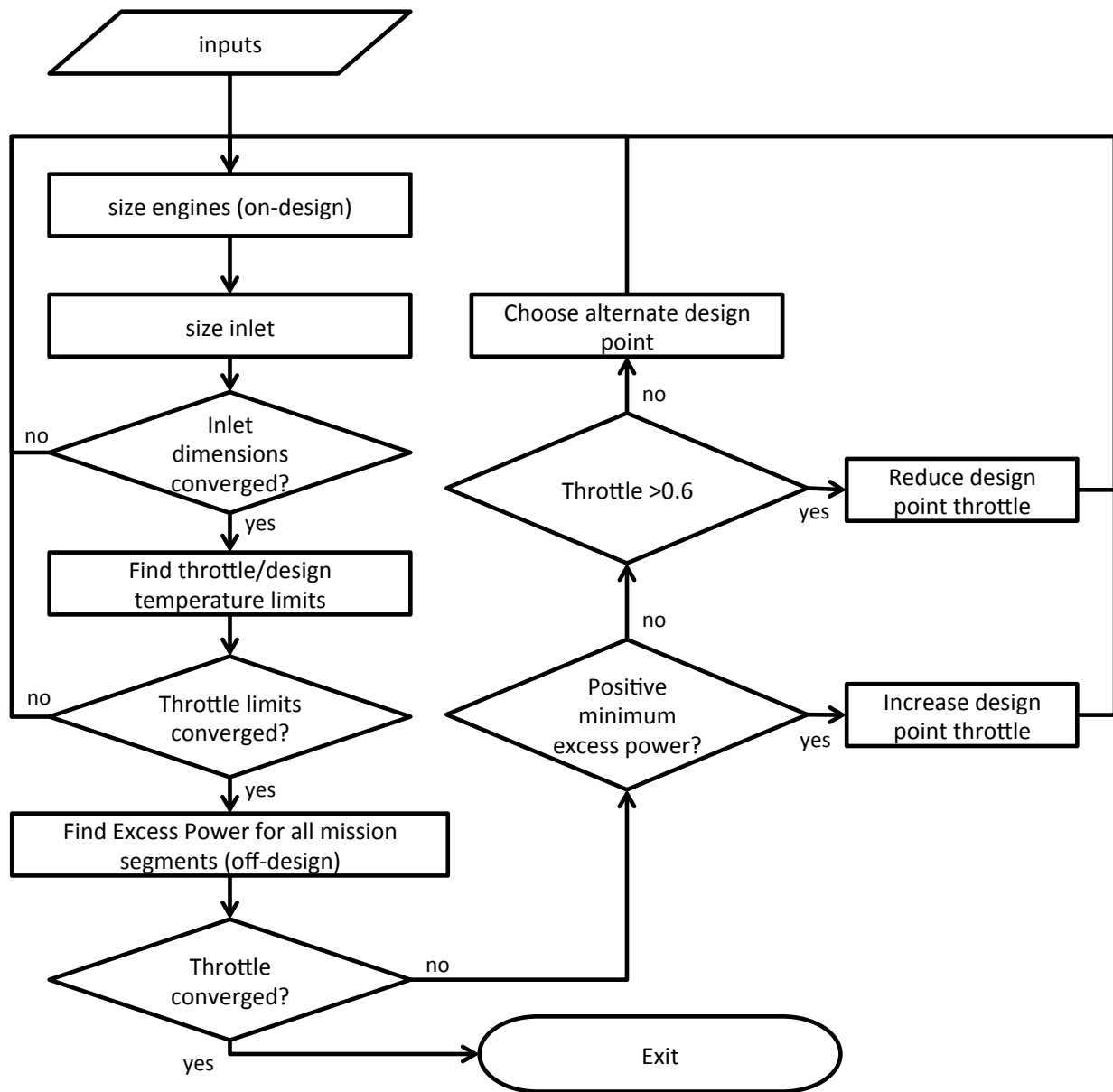


Figure 10.3: Flowchart of the process followed to size the engines

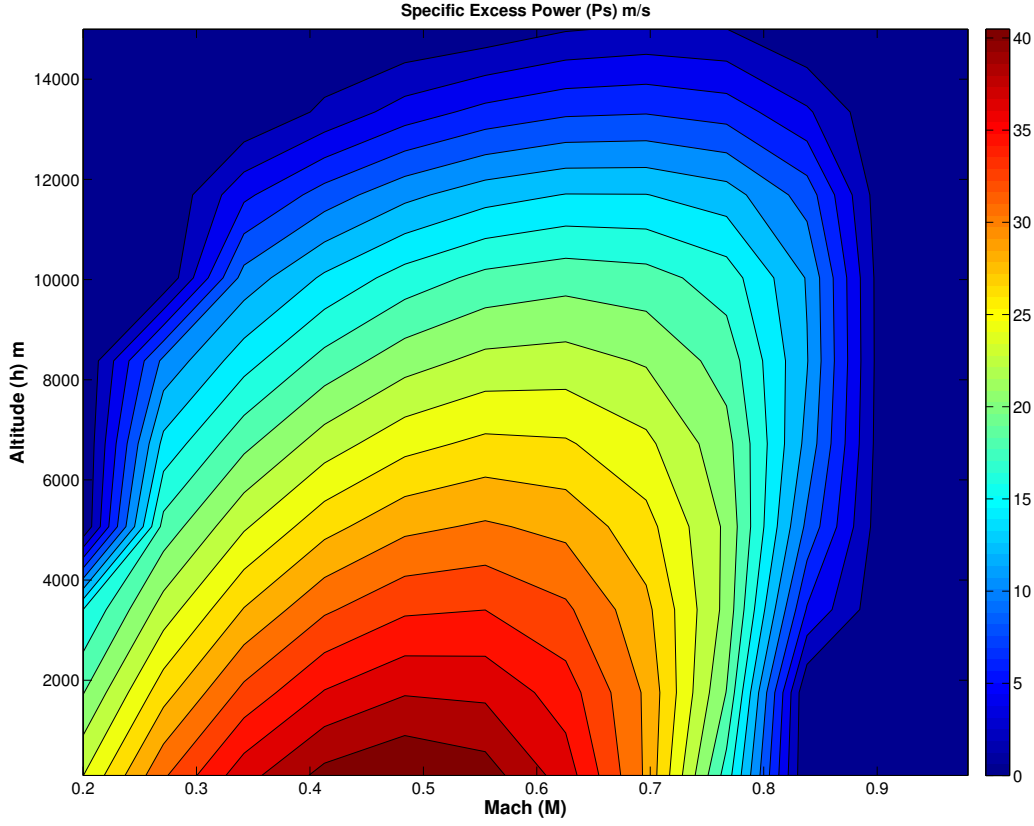


Figure 10.4: Contour plot of specific excess power with respect to altitude and Mach number for a sample aircraft at mid cruise

aircraft the Breguet range equation is used.

$$R = -\frac{aM_\infty}{g \text{ sfc}} \frac{L}{D} \ln(WF_i) \quad (10.6)$$

where the aircraft's L/D refers to the trimmed aircraft value and the engine sfc is found for a throttle setting such that the x-component of installed thrust equals the drag. Both these values are estimated for the mid-segment aircraft weight. To minimise trim drag, by redistributing the fuel load, the longitudinal centre of gravity is moved as close to the aircraft's neutral point as possible, while abiding by the minimum predefined static margin requirements. To improve the accuracy and account for the reduction of aircraft weight as fuel is consumed, cruise mission segments are split into multiple smaller segments. About twenty are used in the present implementation, however this number can easily be altered. The total range is found by summing up all the individual segment ranges found, given by (10.6) in meters.

The same procedure is followed to estimate the aircraft endurance for loiter segments (in seconds).

The equation used is:

$$E = -\frac{1}{g} \frac{L}{sfc} \frac{D}{D} \ln(WF_i) \quad (10.7)$$

A range credit is added for the aircraft's climb and descent segments, considering the average rate of climb and the starting (1) and ending (2) cruise velocities. The added range is therefore:

$$R = \frac{\Delta h(a_1 M_1 + a_2 M_2)}{2(dh/dt)} \quad (10.8)$$

where Δh is the altitude gain or loss. An average climb rate (dh/dt) of $12.7m/s$ and an average rate of descent of $7.6m/s$ are assumed.

The process is repeated for varying amounts of fuel and payload to find the aircraft's payload vs. range envelope. The conditions considered are:

- Maximum payload (W_{pld}) and normal fuel load (W_f)
- Maximum fuel load ($W_{\max fuel}$) and maximum allowable payload ($W_o - W_{\max fuel}$)
- Maximum fuel load ($W_{\max fuel}$) and no payload

The aircraft's ferry range can also be determined by evaluating the case of maximum fuel and no payload while allocating all the fuel available for normal operations to a single cruise segment.

10.4 Takeoff & Landing Distance

The aircraft's takeoff distance is estimated using the methodologies presented in section 4.2.3.2. The aircraft's resistance and equilibrium trim setting however are no longer determined by experimental data but using the hydrodynamic methods presented in chapter 8.

The aircraft's aerodynamic properties are re-evaluated using the VLM to account for the effects of extreme ground proximity on the surface's centre of pressure, zero lift angle of attack and Oswald efficiency factor.

The aircraft landing distance is calculated in a similar way. The total landing distance is split into the approach and waterborne segments. The ground distance covered by the glide from an obstacle height h_{OBS} is given by:

$$S_{GL} = \frac{V_{appr}^2 - V_{td}^2}{2g\bar{\gamma}} + \frac{h_{OBS}}{\bar{\gamma}} \quad (10.9)$$

where $\bar{\gamma}$ is the mean landing glide slope.

$$\gamma = \frac{D - T}{W} \quad (10.10)$$

Roskam suggests that a mean value of approximately 0.1 can be used for $\bar{\gamma}$. The approach velocity is assumed to be:

$$V_{appr} = 1.3 \sqrt{\frac{2W_L}{\rho S_{ref} C_{L_{Lmax}}}} \quad (10.11)$$

and the touchdown velocity is given by:

$$V_{TD} = A_{appr} \sqrt{1 - \frac{\bar{\gamma}^2}{\Delta n}} \quad (10.12)$$

where Δn is heavily dependent on pilot technique and the aircraft's handling qualities. A value of $\Delta n \approx 0.1$ is suggested.

The ground distance covered until the aircraft comes to a full stop is given by the integral:

$$S_G = \frac{1}{2g} \int_{V_{TD}}^0 \frac{dV^2}{a/g} \quad (10.13)$$

where a/g is the instantaneous acceleration found using (4.29). The thrust is null or negative if thrust reversers are available and the values of C_L and C_{D_o} used reflect the deployment of lift dumpers or air brakes, where available. The hydrodynamic resistance to load ratio (R/Δ) is found for the instantaneous equilibrium trim angle, as was done for the takeoff length estimation.

The final landing field length requirement, for an FAR-25 aircraft and an obstacle height of 50ft is given by:

$$S_{FL} = \frac{S_G + S_{GL}}{0.6} \quad (10.14)$$

The takeoff and landing computations can be repeated for varying elevator deflections, in order to identify what deflections are necessary to minimise the takeoff or landing distance and produce trim tracks that fall within the porpoising stability limits for the hull in question.

Chapter 11

Methodology Implementation

11.1 Full Synthesis Implementation

The methodologies presented in the preceding chapters were all collected and implemented within a single aircraft synthesis algorithm. This section presents the architecture of the overarching program, intended to produce an accurate representation of the performance and dimensions of an aircraft design based on a number of inputs.

The procedure followed to obtain a final aircraft design can be seen in figure 11.1. It shows the rather logical sequence in which the aircraft is sized and its performance is evaluated. The synthesis starts with the initial sizing process, returning the aircraft's maximum takeoff weight, wing loading and thrust to weight ratio based on assumed values for the aircraft's lift to drag ratio and stall characteristics. The engines are subsequently sized using the process illustrated in figure 10.3.

The body shape necessary to meet cabin and cargo packaging criteria and the hull length needed to provide sufficient buoyancy are subsequently found. The propulsion system, fins, control surfaces and leading edge devices are then placed at predefined positions relative to the wing geometry. The process described in section 6.2 is used to package the required amount of fuel. The hull's empty weight is then estimated, resulting in a recalculation of the minimum fuel weight that must be packaged as:

$$W_f = W_o - W_e - W_{pld} \quad (11.1)$$

The process is repeated until the minimum necessary fuel volume and empty weight values converge. Having found the empty weight and balance characteristics, the centre of gravity locations under different loading conditions are then found.

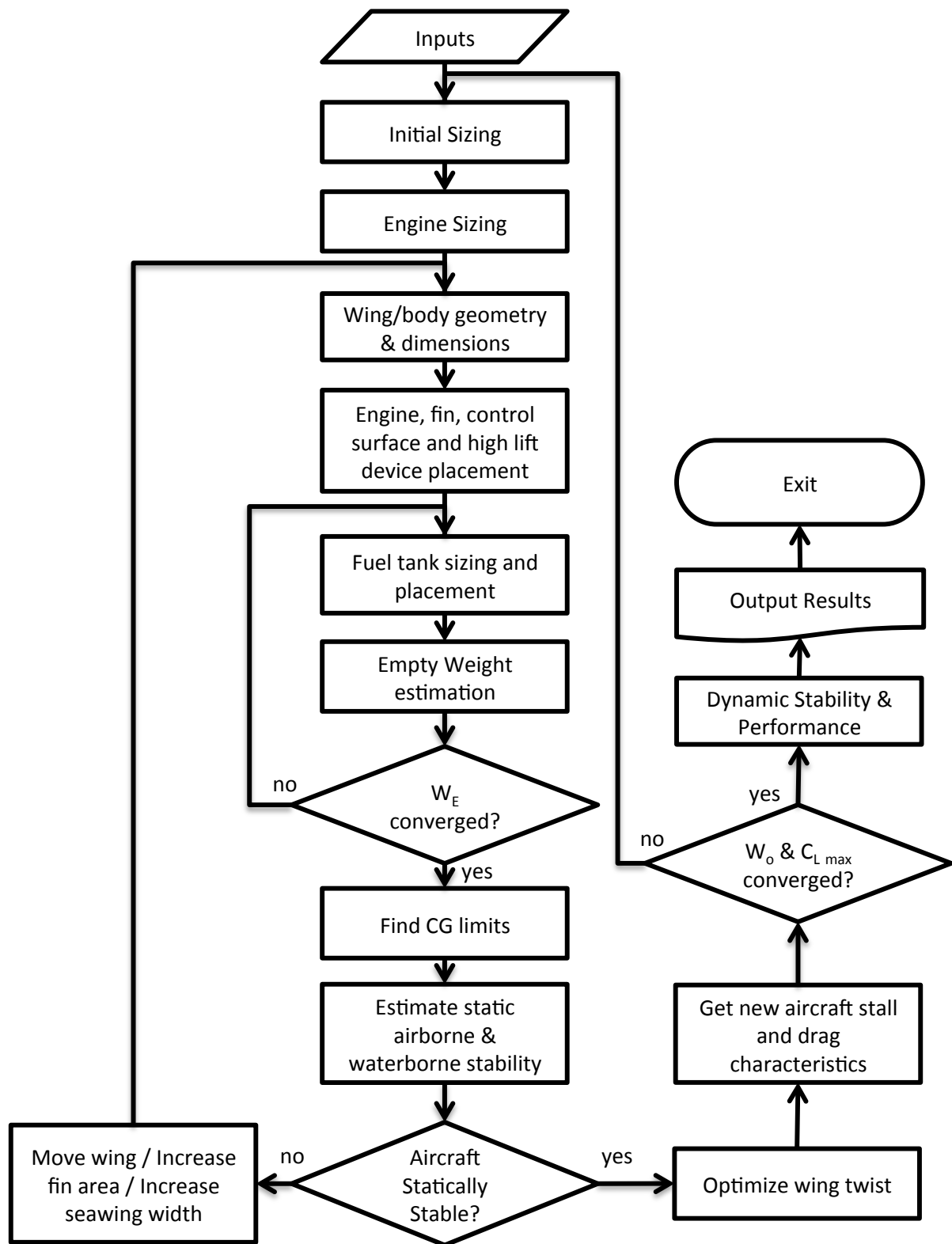


Figure 11.1: Overall synthesis flowchart

The longitudinal and lateral static stability of the aircraft in flight and when at rest on the water is then estimated. If the static margin is found to be excessive, the wing is moved aft by increasing x_w . The wing is moved forward if the static margin must be increased. If the aircraft is found to be laterally unstable, the tip fin aspect ratio is increased until a maximum value is reached. If higher levels of lateral static stability are necessary, additional fins are placed along the aircraft's trailing edge. The aircraft's metacentric height and loll angle are found for both empty and maximum takeoff weight conditions. If the loll angle exceeds the maximum loll angle allowed, the seawing width is increased.

The entire design process, from getting the body shape onwards, is repeated until the aircraft shape and dimensions converge. Typically, the aircraft's high lift devices are sized so that the optimum maximum lift coefficients found during initial sizing are reached. This is easily done for aircraft with horizontal tailplanes as they can counteract the resulting pitching moments. For this particular aircraft however we are constrained to the use of leading edge devices and therefore an iterative approach must be taken until the aircraft's stall characteristics converge to a single value.

To obtain the aircraft's stall characteristics, the wing's twist distribution must first be estimated. Also estimated are the aircraft's profile drag and oswald efficiency for varying cruise mach numbers and aircraft configurations. Using the updated values for empty weight, maximum lift and drag, a better estimate of maximum takeoff weight, wing loading and thrust to weight ratio can be obtained ahead of the next iteration. To ensure convergence, the initial lift to drag assumption is still used by the weight estimation algorithm, while the empty weight is no longer based on an empirical relation but rather on a combination of the previous initial sizing output and the detailed weight estimation result.

Once both the maximum takeoff weight and maximum lift coefficient values converge to within a predefined tolerance, the aircraft's dynamic stability and performance can be evaluated. Looking forward to the code being used within an optimiser, the user is given the ability to bypass the specific excess power and the range vs. payload calculations to reduce the code's running time.

Presently, the typical runtime is between 10-20 minutes using a 2.2 GHz processor and 8 GB of available RAM memory. Increased payload weight or range requirements tend to lead to more overall iterations being necessary, as the initial empirical empty weight estimate seems to be excessive.

11.2 Review of Computational Implementation

11.2.1 Main Synthesis Program

The synthesis algorithm was implemented in FORTRAN 95. This particular language was chosen over the relative ease and simplicity of MATLAB due to the expected large scale use of iterative methods. The advantage of being able to parallelize computationally intensive modules of the program in the future, using OpenMP, also played a role in the decision process.

The program is split into modules, roughly along the same lines as this thesis, each containing the relevant functions and subroutines. Each function only performs a specific task, allowing it to be called by several other functions that require the same task be performed. Other than reducing the coding workload, changes made apply to all relevant methodologies, reducing the likelihood of discrepancies. To further simplify coding, all calculations were done using SI units, unless specifically stated otherwise, thus avoiding mistakes due to the use of erroneous units rather common in aircraft design.

Text files, easily read and written by the user are used for input, output and storage of some intermediate results. All input/output files are named based on a filename and extension (fname) defined by the user at the start of the program. This filename input is the only interactive segment of the code, which subsequently runs without any user inputs. The input files that must be predefined are:

inpt-fname Input file containing major design parameters and packaging constraints. It can be generated by a separate input program to simplify input.

mission-fname File containing all design specifications such as mission profile, passenger numbers, cargo weight, service ceiling, approach speed constraints and field length.

SFC-fname Assumed engine specific fuel consumption value. Either a single value can be provided or a lookup table for varying thrust level, altitude and cruise mach number, originating from an existing engine design.

cabinconst-fname File containing the cabin packaging constraints, necessary to find the size and weight of the passenger cabin.

cargodim-fname File containing the dimensions of the cargo container to be used.

Lasmd-fname Initial assumptions for the maximum lift coefficient of the aircraft in clean, takeoff and landing configurations.

Dasmd-fname Initial assumption for the aircraft's drag characteristics in clean, takeoff and landing configurations. It can contain as single lift to drag ratio assumption, distinct L/D values for cruise and loiter or profile drag and Oswald efficiency values for different aircraft flight configurations.

hullairf-fname Parametric cross-sectional shape definition for aircraft centre section ($y < y_{le}$).

wingairf-fname Parametric cross-sectional shape definition for the aircraft's outer wing section.

flaps-fname File describing the type, dimensions and maximum deflections of leading and trailing edge high lift devices to be used.

Updated values for the aircraft's drag and stall characteristics, necessary for the initial sizing process, obtained after the first iteration are saved in the files where the predefined filename is preceded by the letter 't', indicating it is a temporary output. The fact that the initial sizing process subroutines must repeatedly read data from files does result in a sizeable time penalty and therefore in future versions this information should be only read once into program variables, which can be subsequently updated.

All important information generated during the synthesis process is stored into subject specific structures. Using a separate structures to store all aerodynamic coefficients or all weight and balance information, coding is simplified and access to individual pieces of information are easily retrieved. Arrays of structures are also used for information that is evaluated at multiple points, such as the aircraft's cross sectional shape variation along the wing's span. For this particular information, the shape was defined using 200 points to describe the upper or lower surface at 151 semi-spanwise stations. This rather high total number of ordinates was necessary to allow the accurate representation of the aircraft's aerodynamic shape, while not radically increasing the computational cost of the numerous bilinear interpolations necessary throughout the design process.

Several numerical methods were used to identify optimum points and solve nonlinear functions encountered throughout the synthesis. The initial sizing algorithm utilises the bisection method to find the optimum combination of thrust to weight ratio and wing loading for varying values of takeoff and landing maximum lift coefficient, taken between the clean aircraft $C_{L_{max}}$ and the predefined maximum values. The same method is also used to estimate the engine's specific fuel consumption

at the required thrust output, as the OFFX algorithm often returns NaN values for excessively low throttle settings.

Empirical methodologies, as presented in print, often require that values be read from the graphs provided. When these graphs could not be accurately represented using a simple equation thus reducing computational effort, the graphs were digitised using a sufficient number of data points and linear interpolation/extrapolation was used.

Ideally the Newton-Raphson method would be employed to find the optimum wing twist distribution, however the resulting twist distributions were exceptionally peaky and only met the desired lift distribution at the control points, leading to an increased level of induced drag. Attempts to use the same numerical method to minimise the sum squared lift distribution error for all wing increments showed very poor convergence characteristics. Considering the time taken for each VLM case to be run and the exponential impact of increasing the number of control points, an iterative method was opted for instead.

The need to repeatedly use the VLM though out the synthesis to estimate the aircraft's aerodynamic characteristics results in a substantial computational time penalty. To improve the VLM code's efficiency, operations were vectorised where possible. The wing's vorticity distribution is found by solving the vectorised system of linear equations using the Gauss-Jordan method. Matrix inversion was also extensively used to solve the parametric airfoil equations in order to obtain the upper and lower surface polynomial curve coefficients. This particular computational method was chosen for its robustness in inverting matrices of all types. The Gauss-Jordan method is by no means the most efficient way of inverting a matrix, therefore the use of an alternate algorithm should be considered for future implementations. MATLAB would have been substantially more efficient in solving the systems of equations, however it is expected that a higher level language implementation is more efficient overall, given the number of additional computations necessary to obtain a result. Ultimately the best way to reduce the computational time impact of the VLM is to use the minimum number of panels that still produce accurate results. A study performed on both an elliptical and a 30 degree backswept straight tapered wing section showed that accurate results could be obtained using a matrix of 25 spanwise by 7 chordwise panels. An additional mesh of 5×7 panels minimum is also added for each fin modelled. The number of panels must obviously be doubled for non symmetrical cases, greatly impacting the code's running time. Consequently lateral airborne static stability checks were only performed once the aircraft had been found to be longitudinally statically stable and the amount by which the fin area was increased was of the order of 10-20% rather than the typical increments of 0.1-2% used in the longitudinal case.

To evaluate the aerodynamic effects of the step, the aerodynamic properties of several sections must be evaluated. This is done by using the executable binaries for both Xfoil and VGK. The program was designed to operate in a UNIX environment and will therefore work with both Linux and OS X systems. A platform appropriate build of Xfoil is therefore necessary. On the other hand, VGK is only available as a DOS executable and therefore the prior installation of the WINE open source emulation package is necessary. To ensure the executables can be found, their paths must be defined prior to compiling the code by altering the appropriate string variables. Input and output from both programs is done using a text file which contains the airfoil coordinates and batch files (.bat) used to automatically input all commands that typically involve user interaction.

The airfoil coordinates inputted were often found not to produce smooth airfoil distributions in Xfoil, likely due to the resulting panelling. Although these fluctuations in pressure have little effect on the predicted values for lift and moment, they have a grave effect on the convergence of the iterative viscous-inviscid matching algorithm used to estimate viscous drag. The convergence issue is exacerbated at high Re numbers and moderate angles of attack.

The empty weight and maximum lift coefficient values used for each iteration of the synthesis are damped to ensure convergence. The empty weight, $(W_e)_i$, used for the i^{th} initial sizing iteration was therefore:

$$(W_e)_i = (1 - a)(W_e)_{est} + a(W_e)_{i-1} \quad (11.2)$$

where $(W_e)_{est}$ is the empty weight found by the preceding detailed weight estimation. Without the use of a damping factor (a), the 2.5% absolute percent error convergence requirement for maximum takeoff weight and 10% requirement for clean aircraft $C_{L_{max}}$ would not be met. Values of $a = 0.1$ for W_e and $a = 0.5$ for $C_{L_{max}}$ were found to produce a reasonable combination of accuracy and number of iterations necessary for convergence.

To further ensure that computational time is not substantially increased for fractional increases in accuracy, a maximum number of iterations was set. A maximum number of 15 wing placement/static stability iterations was therefore allowed. The total number of times the entire aircraft sizing process could be run was also limited to 16 iterations.

11.2.2 Graphical Output

Figures showing the aircraft's overall shape and packaging as well as plots detailing its performance characteristics are an easy way for the user to evaluate the resulting designs. As FORTRAN, like most other high level programming languages, does not have graphical output capabilities, the synthesis code produces test files containing the aircraft's geometry and performance characteristics. These can then be read by any number of programs capable of producing figure and plots, The scientific programming package MATLAB was chosen for its ability to easily produce two-dimensional, contour and surface plots, the majority of which can be seen as sample aircraft outputs throughout this thesis. The output files generated and used to produce output figures are:

airplaneout-fname Coordinate file used to describe the aircraft's entire three-dimensional shape and containing all packaged system dimensions. It does not contain coordinates for the engine pylons and shows the aircraft with an unfaired step. To reduce the file size, as all aircraft designed are symmetrical, the shape of only half the aircraft ($y \geq 0$) is given.

packaging-fname Simple output, not used for the production of any figures, that details the packaging of the cabin and cargo bays. It also provides information on the maximum number of passengers or cargo containers that can be accommodated within each bay.

crpt-fname Output containing the carpet plot used to determine thrust to weight ratio and wing loading, at the optimum maximum C_L values found. Unlike all other outputs. it is optimised for easy plotting using Gnuplot but with the appropriate data input algorithm can be used with any plotting tool.

Tinst-fname Performance output, showing the maximum installed thrust in Newtons that the aircraft can produce at varying altitudes, given in meters, and mach numbers.

Tuninst-fname Same as the installed thrust file but for the total thrust when intake losses are not considered.

LtD-fname Variation of the aircraft's trimmed Lift-to-Drag ratio with altitude and mach number, evaluated at mid cruise weight. Very low values are returned when the aircraft cannot be trimmed or has stalled.

Psplot-fname Data necessary to produce the aircraft's P_s plot at mid cruise. Values where the aircraft is stalled or could not be trimmed are given as excessively negative.

AoA-fname File containing the trimmed angle of attack relative to the aircraft baseline, in degrees, of the aircraft for varying altitude and mach number, evaluated at mid cruise. A null value is returned when the aircraft cannot be trimmed or is stalled.

deltae-fname File containing the elevator deflection necessary to trim the aircraft for varying altitude and mach number. Trim is evaluated at mid cruise for a longitudinal centre of gravity position as close to the minimum static margin as possible. A null value is returned when the aircraft cannot be trimmed or is stalled.

LtDnt-fname Variation of the aircraft's untrimmed Lift-to-Drag ratio with altitude and mach number, evaluated at mid cruise weight. It is used to easily illustrate the amount of trim drag generated by the control surface deflections necessary for trimmed flight.

attdstat-fname File indicating the aircraft's equilibrium position when at rest on water. It gives the centre of gravity position for an empty and fully loaded aircraft as well as the height of the CG above the waterline and trim and loll angles. It is used to ensure that parts of the aircraft that should not be wetted aren't.

Takeoff-fname & Landing-fname Files containing the computed takeoff and landing distances of the resulting aircraft design. They also include the trim tracks followed during the takeoff and landing manoeuvre for different elevator deflections and the predicted upper and lower porpoising stability trim limits.

Also outputted is the paneling used for the VLM code (VLMPANELS.dat) and the aircraft's lengthwise cross sectional area distribution (AreaDist.dat) for the entire aircraft and the aircraft structure with and without the fins.

Chapter 12

Case Study

The aircraft design and performance given by the synthesis algorithm described in chapter 11 depends heavily on the prescribed mission profile and a number of geometrical input variables. The results of two sample cases are given in this chapter.

Section 12.1 presents a sample 300 passenger, medium range airliner. The resulting design is the result of a prompt manual optimisation process. A number of parameters were chosen based on past seaplane hull designs and only a handful of parameters were subsequently altered until the aircraft's MTOW was roughly minimised and the required cruise flight conditions could be met. The variables altered were the reference wing aspect ratio ($\mathcal{R}$) and quarter chord sweep ($\Lambda_{c/4}$), as well as the hull length to beam ratio (L/B) and the assumed magnitude of cruise lift to drag ratio, given by $(L/D)_{in}$.

The main driver of aircraft weight was the assumed lift to drag ratio, used in the initial design calculations. If too high a value was assumed, the resulting aircraft was unable to complete the pre-defined mission profile. Alternatively, if too low a value was used, the aircraft's range was higher than necessary and the MTOW was substantially increased. The hull length to beam ratio was altered such that the centreline thickness to chord ratio remained reasonable, ensuring adequate transonic aerodynamic performance, and until the weight was minimised. The reference wing geometry, which has an effect on both the final aircraft planform and the aircraft drag divergence Mach number, was finally altered if necessary.

A similar procedure was followed for the sample case in section 12.2, showing a 900 passenger, long-range airliner. The design shown is not optimised to the extent of the previous sample output, as it is mostly intended to show the synthesis algorithm's flexibility in dealing with designs featuring

multiple floors. Further manual optimisation was found to produce a single floor layout, of higher length to beam ratio.

12.1 300 Passenger, Medium-Range Aircraft

A sample 300 passenger, medium range airliner design is presented in this section. The mission profile to be fulfilled was defined based on that seen in figure 4.3 with a cruise range of 6000 km at $M = 0.7$ and a loiter endurance requirement of 2500s at $M = 0.5$, both at an altitude of 35000 ft. A takeoff/landing distance constraint of 2.0km was applied and a maximum landing to takeoff weight ratio of 0.8 chosen. The aircraft was also required to be capable of reaching a maximum Mach number of 0.8 at cruise altitude. A minimum approach speed constraint of 140 kts was also applied.

The main aircraft and hull shape input parameters used are:

$\mathcal{R}$	10.0	λ	0.3	$\Lambda_{c/4}$	35.0
$2y_{le_{\max}}/b$	0.45	$2y_{te}/b$	0.55	$2dy_s/b$	0.20
$2y_{c_{le}}/B$	0.50	$2y_{c_{te}}/B$	0.60	L/B	7.5
N_{st}	1	x_{st}/B	0.10	L_{flat}/L	0.30
δ_k	8.0	h_1/B	0.085	δ_s	90.0
β_1	20.0	β_{ab}	21.0	γ_{hr}	65.0
BPR	8.0	$(L/D)_{in}$	16.2	$(x_{sm}/\bar{c})_{\min}$	0.0

Table 12.1: Major input design parameters for a sample 300 passenger, medium range aircraft

A three dimensional rendering of the resulting aircraft shape can be seen in figures 12.1 and 12.2. The wing planform found to provide the necessary magnitude of static margin and the layout of all major systems can be seen in figures 12.3 and 12.4. Also shown in this figure is the centre of gravity position for an empty aircraft (o) as well as the most extreme forward and aft CG position predicted.

The cross sectional shape variation of the aircraft and the effect of blending the hull into the wing on the airfoil shapes is given in figure 12.5. Finally, the hull lines of the resulting aircraft can be seen in figure 12.6.

The major sizing and aerodynamic parameters given by the sizing algorithm can be seen in table 12.2. It should be noted that all aerodynamic values are estimated based on the reference wing

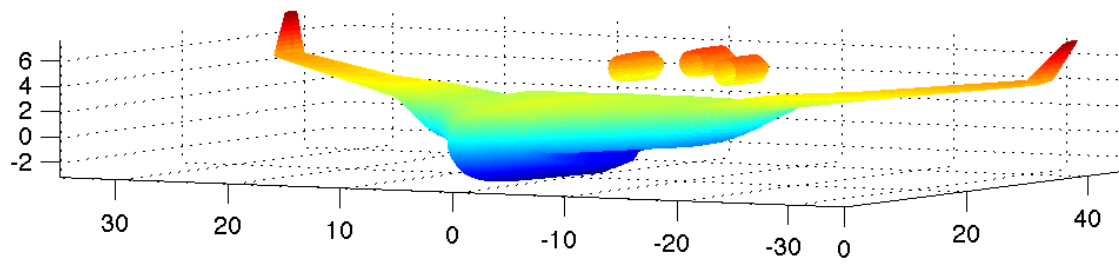


Figure 12.1: Forward side view of the 300 passenger sample aircraft

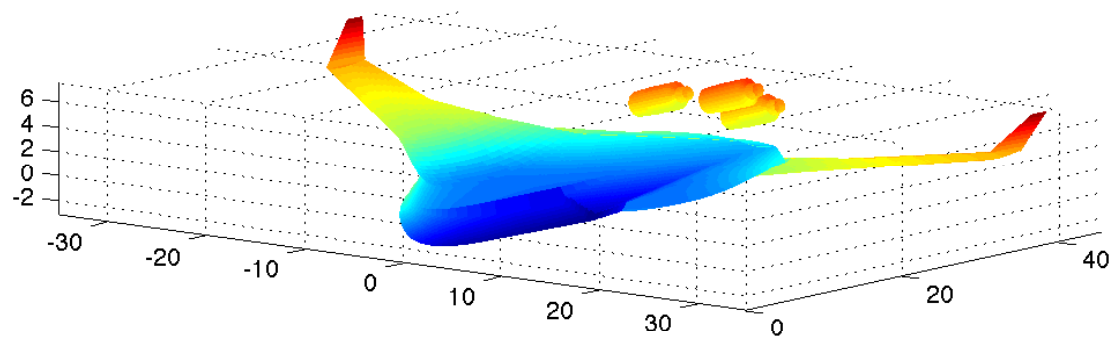


Figure 12.2: Lower rear side view of the 300 passenger sample aircraft

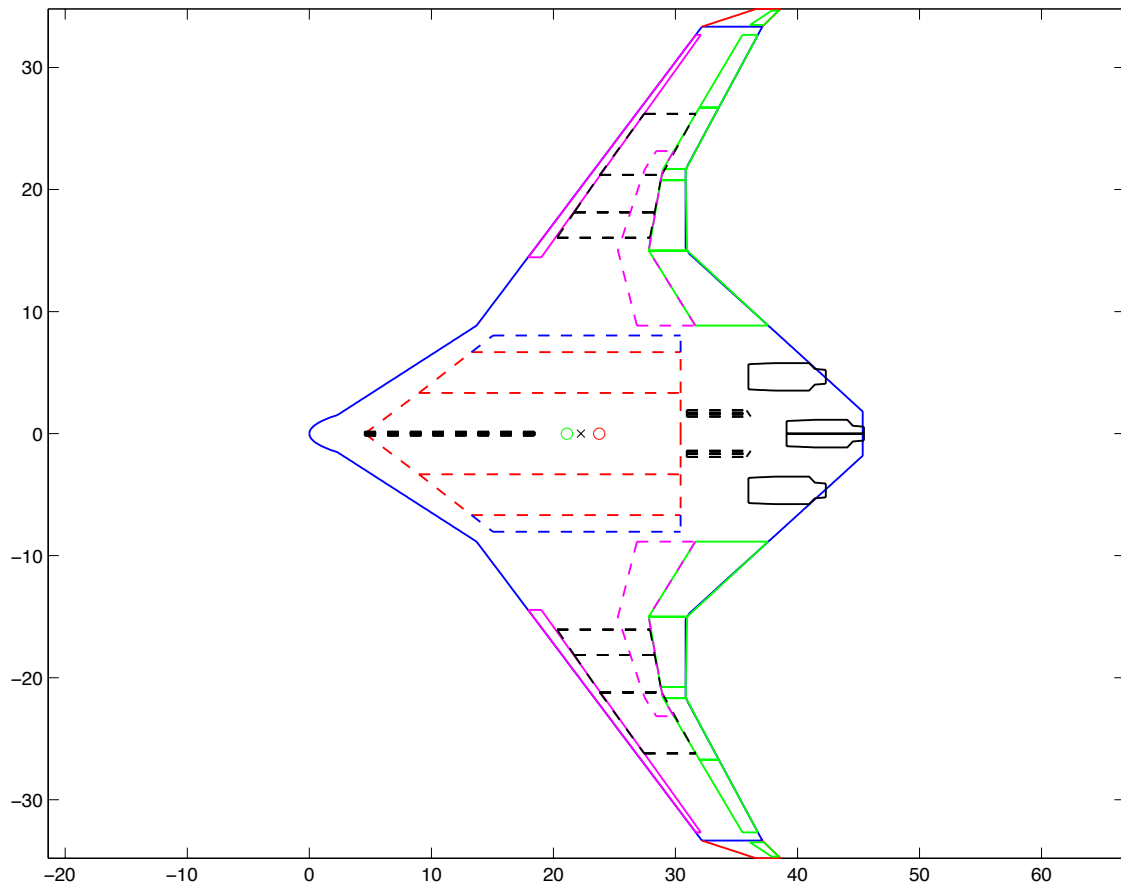


Figure 12.3: 2D Systems layout for a sample 300 passenger aircraft

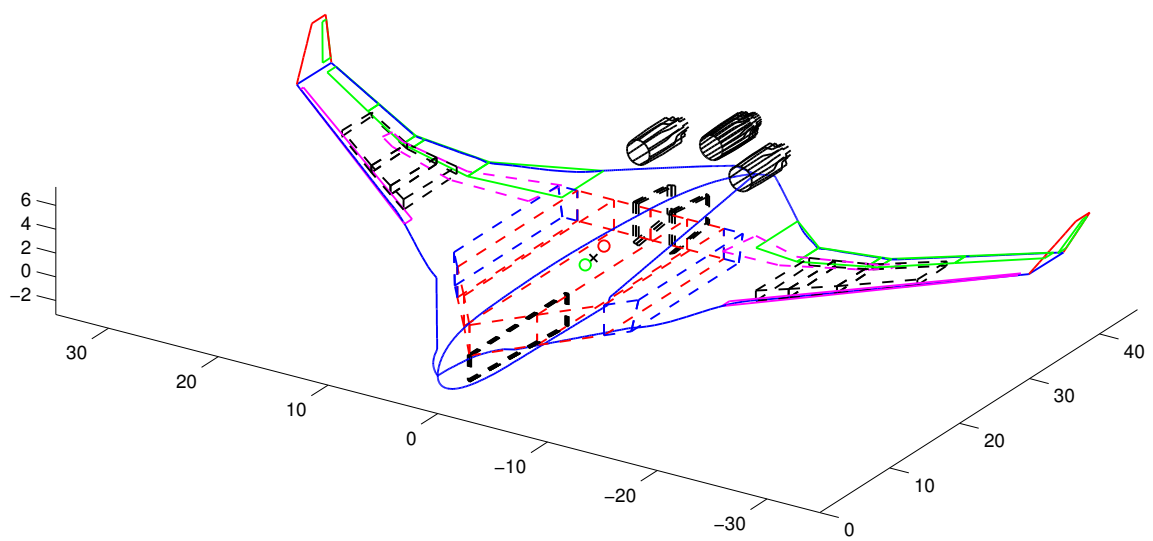


Figure 12.4: 3D Systems layout for a sample 300 passenger aircraft

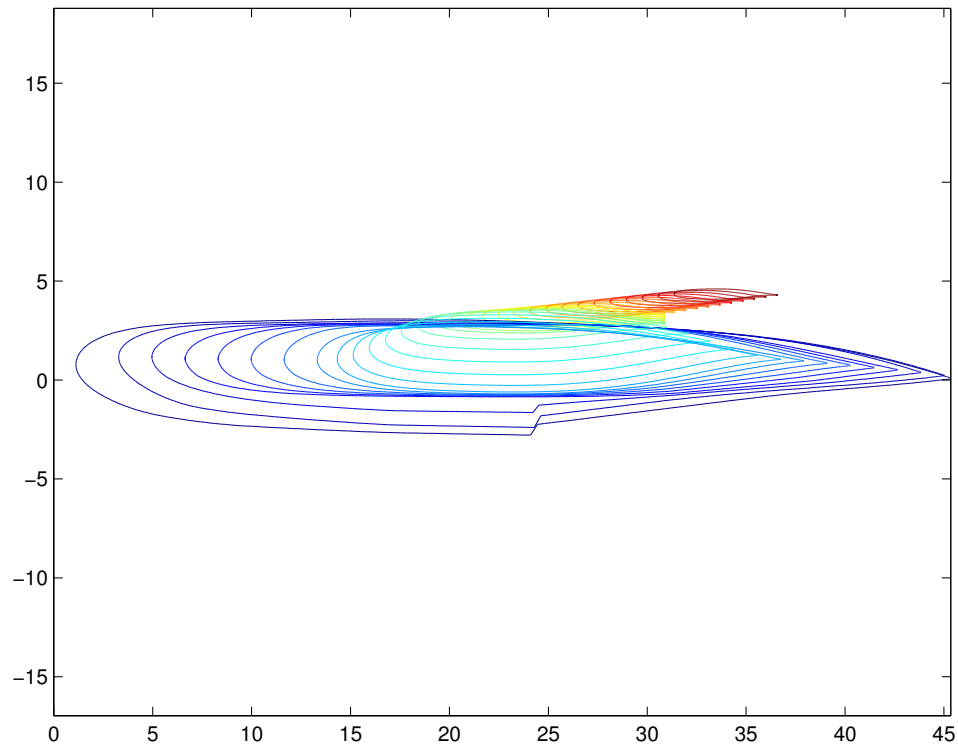


Figure 12.5: Cross sectional airfoil shapes of a sample 300 passenger aircraft

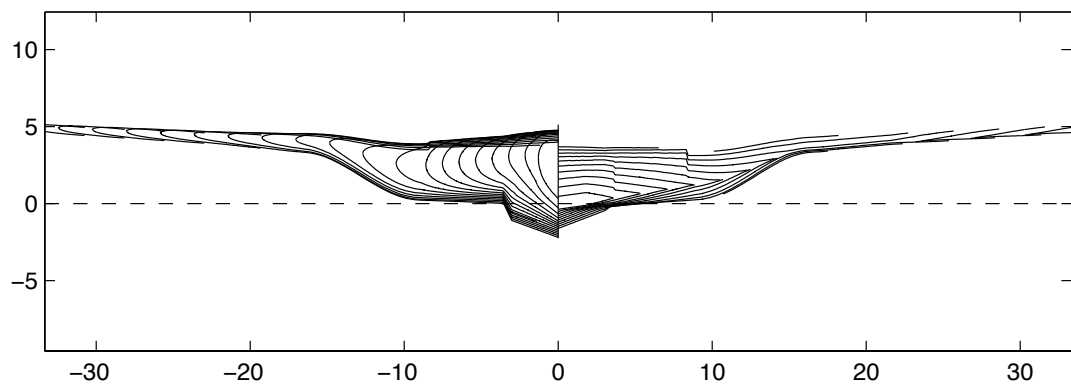


Figure 12.6: Hull lines of a sample 300 passenger aircraft at equilibrium on water at the MTOW

dimensions.

$(T/W)_o$	0.391	W_o/S_{ref}	4096.927 N/m^2	C_{Δ_o}	0.840
L	45.341m	W_o	1822.705 kN	W_e	947.474 kN
W_f	549.164 kN	L_f/B	4.030	$(t/c)_o$	0.140
$(C_{L_{max}})_{clean}$	1.657	$(C_{L_{max}})_{TO}$	1.691	$x_{sm}/\bar{c}$	1.048%
C_{L_o}	0.554	$dC_L/d\alpha$	11.374	$(C_{D_o})_{clean}$	0.01631
$(C_{D_o})_{TO}$	0.02279	$(\Delta C_{D_o})_{step}$	0.00216	e_{clean}	0.920
x_{np}/L	0.526	C_{n_β}	0.176	C_{l_β}	-4.660

Table 12.2: Major outputs for a converged sample 300 passenger aircraft synthesis

The aircraft's range and endurance performance for varying weights is summarised in table 12.3. Figure 12.8 shows the aircraft's trimmed lift to drag ratio, specific excess power, as well as the trimmed aircraft angle of attack and percent control surface deflection necessary. The maximum elevator deflection is set to 25 degrees. The untrimmed aircraft L/D can be seen in figure 12.9. Both these figures are evaluate at the mid cruise weight and lowest possible static margin. The variation of longitudinal CG position with weight and fuel loading conditions can be seen in figure 12.7.

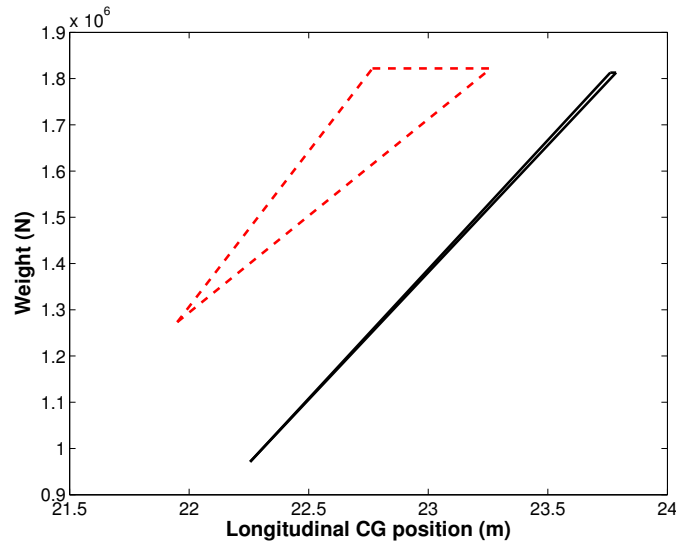


Figure 12.7: CG envelope for a sample 300 passenger, medium-range airliner

The aircraft was found to be statically stable when resting on water both at empty and maximum takeoff weight, with a lateral metacentric height of 0.601m and 5.398m respectively. In flight, the

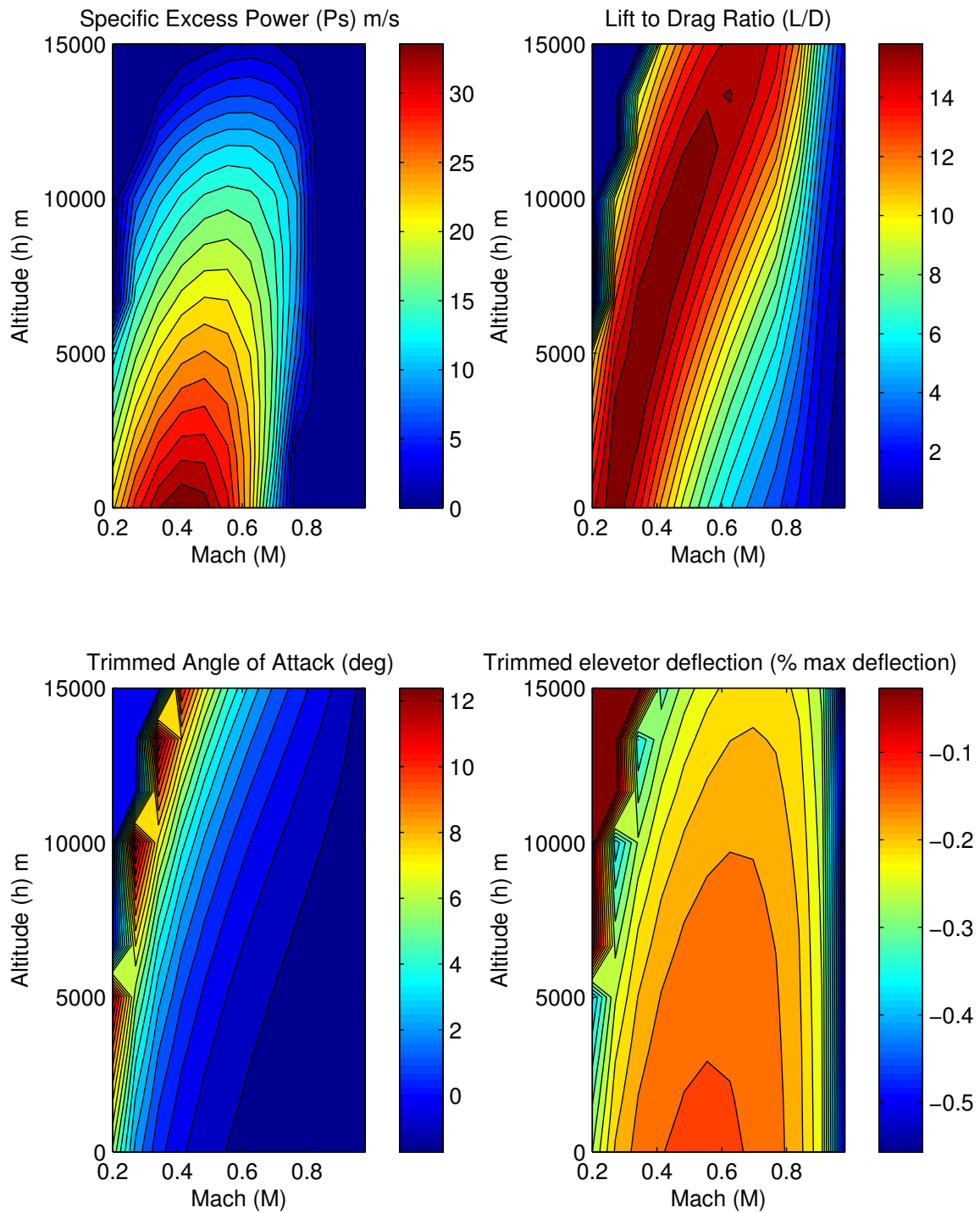


Figure 12.8: Airborne performance & trim characteristics of a 300 passenger sample aircraft at mid cruise

$\%W_{pld}$	Cruise Range (km)	Loiter Endurance (min)
1.0	6164.68	55.22
0.0	9350.01	367.29
0.0	12328.31	-

Table 12.3: Variation of range and endurance with takeoff weight for a sample 300 passenger medium-range aircraft

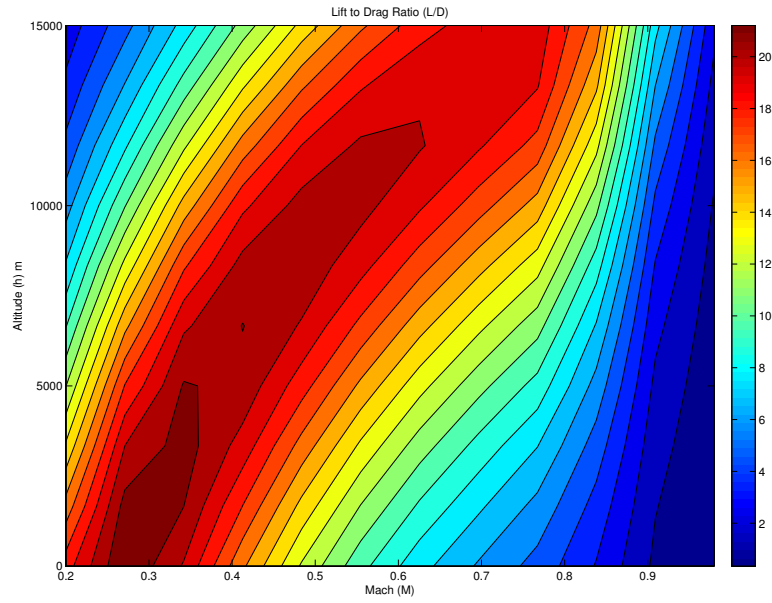


Figure 12.9: Contour plot of the untrimmed lift to drag ratio variation with altitude and cruise mach number for a 300 passenger sample aircraft

aircraft is predicted to be stable in all dynamic modes, which are detailed in table 12.4.

Mode	Natural Frequency (Hz)	Damping Ratio	Time to Double/Half (s)
Phugoid	0.01377	0.04740	169.01
SPPO	0.48454	0.63541	0.36
Dutch Roll	0.20772	0.06095	8.71
Spiral		Stable	

Table 12.4: Predicted frequencies and damping ratios for the dynamic modes of a 300 passenger sample aircraft

Finally the aircraft was found to have a BFL of 1790m and a landing field length requirement of 1413m. Normal takeoffs require 1397m of waterway. The aircraft is expect to not suffer from porpoising during takeoff. The predicted takeoff trim track and the predicted stability limits can be seen in figure 12.10.

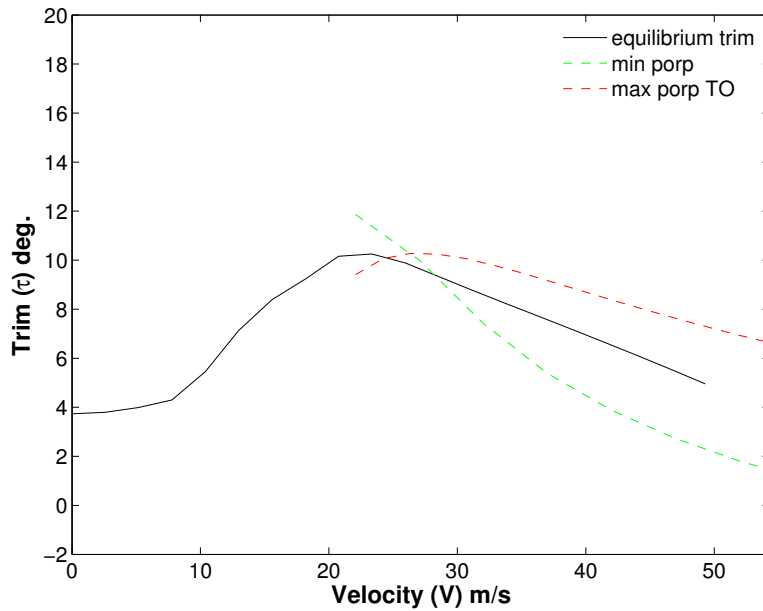


Figure 12.10: Plot of equilibrium trim angle and predicted stability limit variation with velocity during takeoff for a 300 passenger medium-range aircraft

This aircraft lies towards the lower end of the design synthesis' applicability range, in a region where very large length to beam ratios are necessary to maintain a reasonable thickness to chord

ratio. Although the results presented represent anything but an optimised aircraft, the aircraft was found to have decent airborne performance characteristics. The aircraft's mission profile is closest to that of a 757-300. This seaplane's estimated empty weight is about 30% higher than that of the existing aircraft, which however is required to carry only about 250 passengers.

12.2 900 Passenger, Long-Range Aircraft

A sample 900 passenger, long range airliner design is presented in this section. The mission profile to be fulfilled required a cruise range of 15,500 km at $M = 0.8$ and a loiter endurance requirement of 3600s at $M = 0.5$, both at an altitude of 35000 ft. A takeoff/landing distance constraint of 3.0km was required and a maximum landing to takeoff weight ratio of 0.8 chosen. The aircraft was also required to be capable of reaching a maximum Mach number of 0.9 at cruise altitude. A minimum approach speed constraint of 140 kts was also applied.

The main aircraft and hull shape input parameters used are:

$\mathcal{R}$	8.5	λ	0.3	$\Lambda_{c/4}$	40.0
$2y_{le_{\max}}/b$	0.55	$2y_{te}/b$	0.60	$2dy_s/b$	0.2
$2y_{c_{le}}/B$	0.5	$2y_{c_{te}}/B$	0.0	L/B	5.3
N_{st}	1	x_{st}/B	0.1	L_{flat}/L	0.3
δ_k	8.0	h_1/B	0.085	δ_s	90.0
β_1	22.0	β_{ab}	22.0	γ_{hr}	65.0
BPR	8.0	$(L/D)_{in}$	17.6	$(x_{sm}/\bar{c})_{\min}$	0.0

Table 12.5: Major input design parameters for a sample 900 passenger, long range aircraft

A three dimensional rendering of the resulting aircraft shape can be seen in figures 12.11 and 12.12. The wing planform found to provide the necessary magnitude of static margin and the layout of all major systems can be seen in figures 12.13 and 12.14. Also shown in this figure is the centre of gravity position for an empty aircraft (o) as well as the most extreme forward and aft CG position predicted.

The cross sectional shape variation of the aircraft and the effect of blending the hull into the wing on the airfoil shapes is given in figure 12.15. Finally, the hull lines of the resulting aircraft can be seen in figure 12.16.

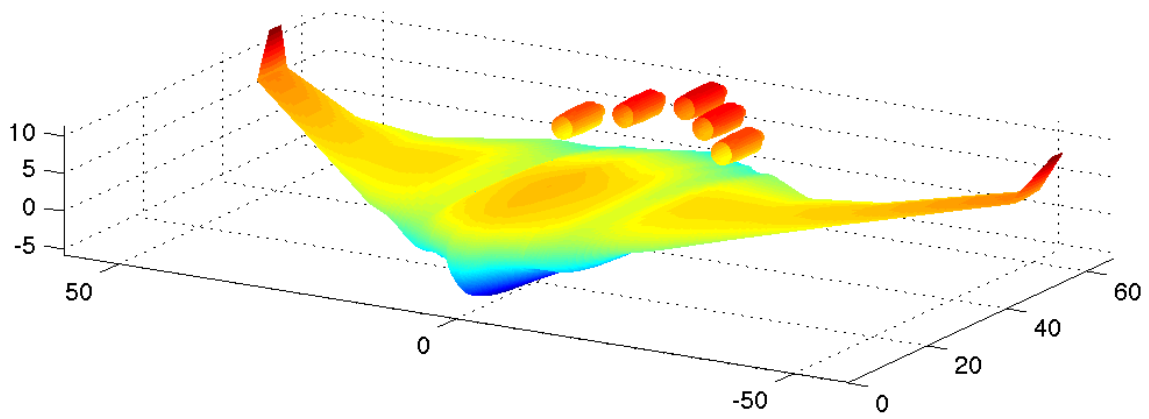


Figure 12.11: Forward side view of the 900 passenger sample aircraft

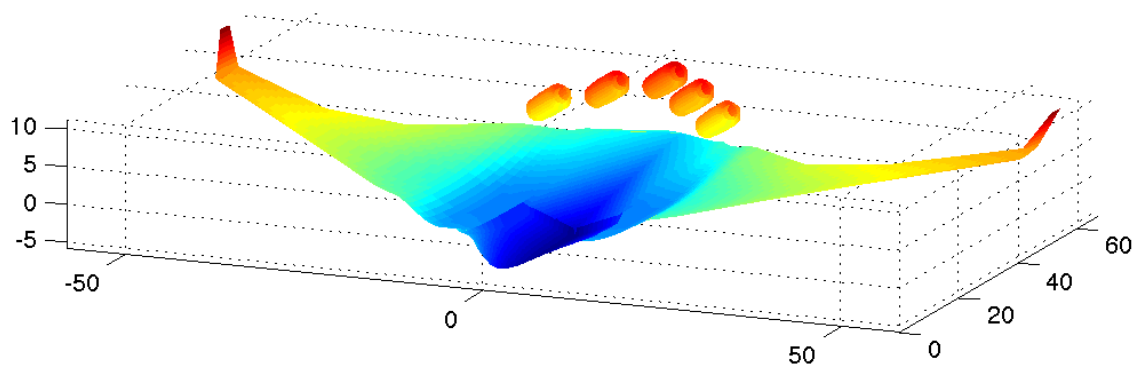


Figure 12.12: Lower rear side view of the 900 passenger sample aircraft

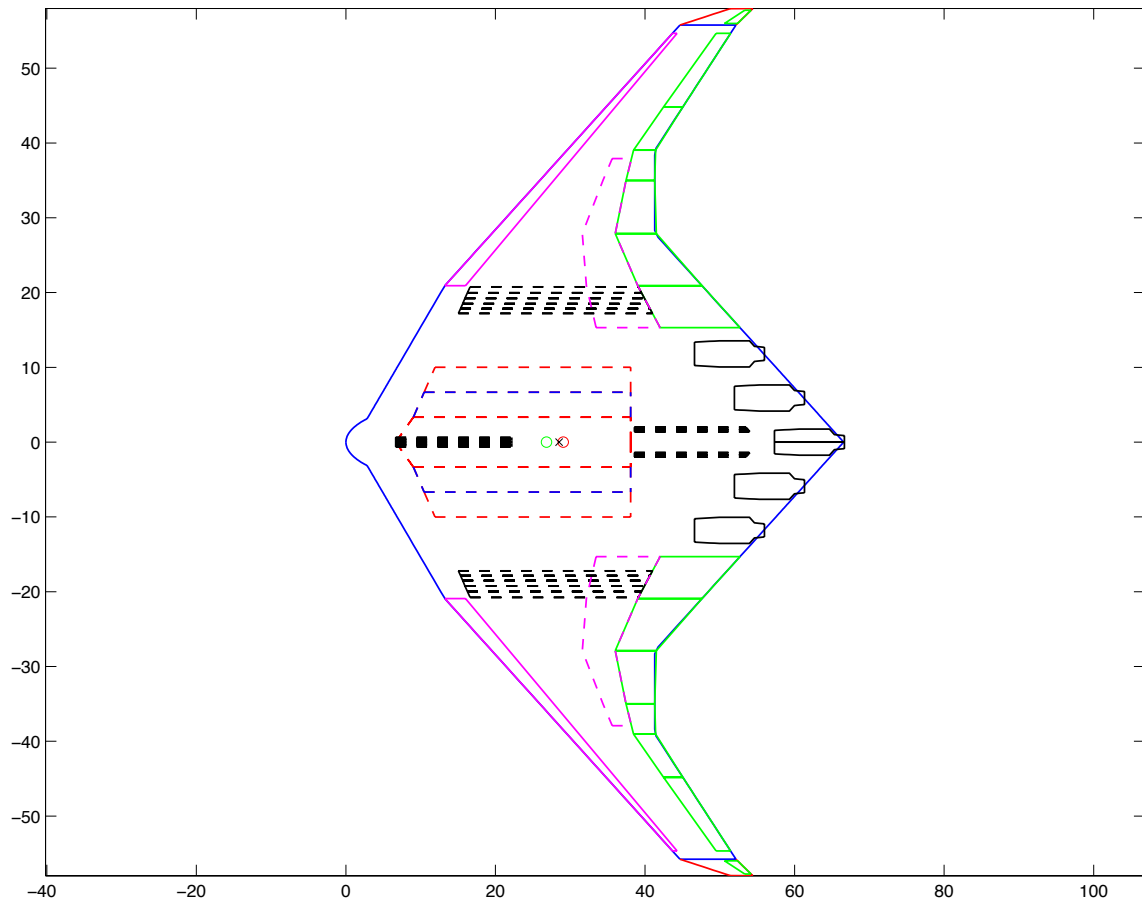


Figure 12.13: 2D Systems layout for a sample 900 passenger aircraft

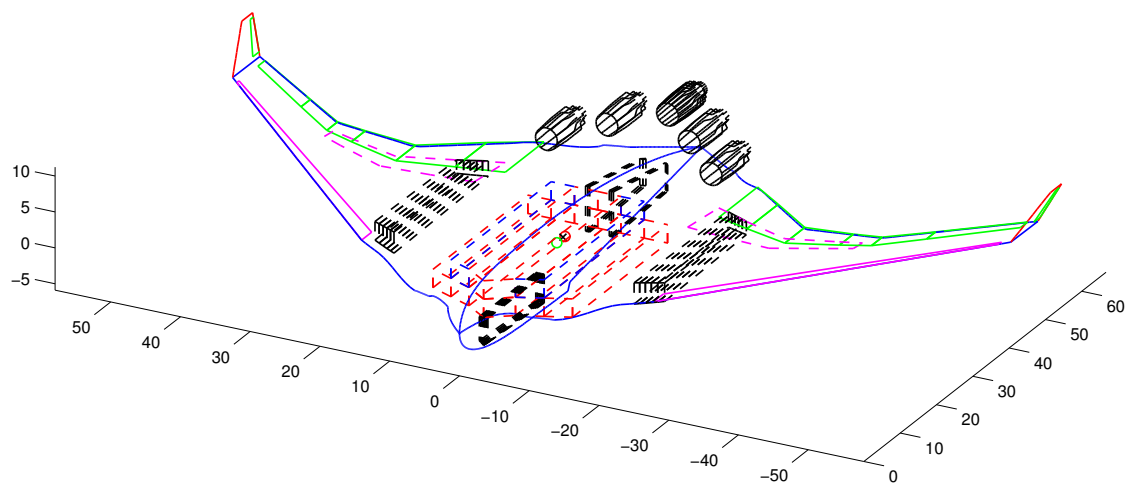


Figure 12.14: 3D Systems layout for a sample 900 passenger aircraft

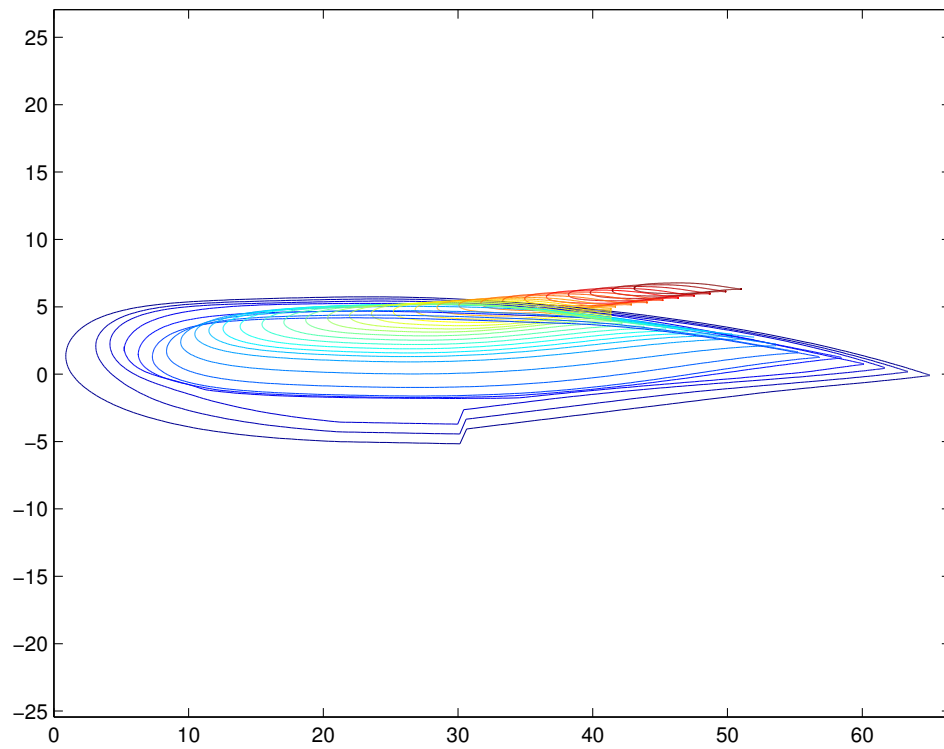


Figure 12.15: Cross sectional airfoil shapes of a sample 900 passenger aircraft

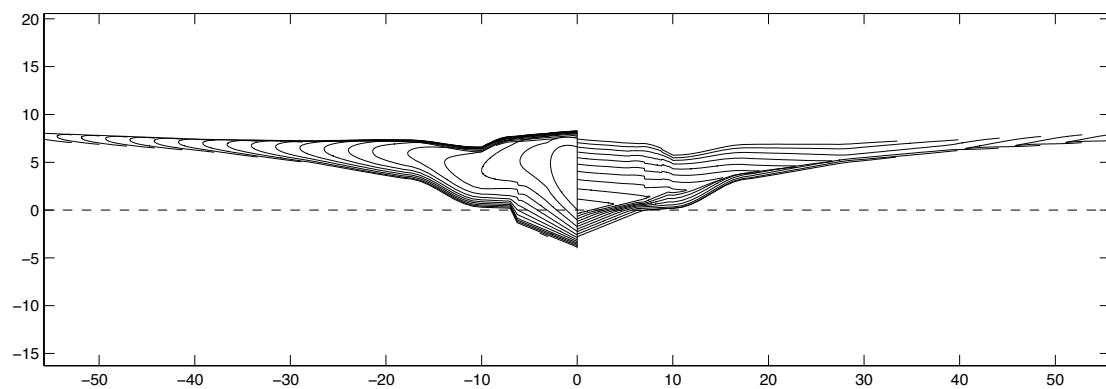


Figure 12.16: Hull lines of a sample 900 passenger aircraft at equilibrium on water at the MTOW

The major sizing and aerodynamic parameters given by the sizing algorithm can be seen in table 12.6. It should be noted that all aerodynamic values are estimated based on the reference wing dimensions.

$(T/W)_o$	0.329	W_o/S_{ref}	5325.846 N/m^2	C_{Δ_o}	0.401
L	66.565m	W_o	7804.451 kN	W_e	3105.583 kN
W_f	3195.657 kN	L_f/B	2.414	$(t/c)_o$	0.177
$(C_{L_{max}})_{clean}$	2.090	$(C_{L_{max}})_{TO}$	1.135	$x_{sm}/\bar{c}$	2.001%
C_{L_o}	0.4216	$dC_L/d\alpha$	10.200	$(C_{D_o})_{clean}$	0.01462
$(C_{D_o})_{TO}$	0.01923	$(\Delta C_{D_o})_{step}$	0.00447	e_{clean}	0.951
x_{np}/L	0.441	C_{n_β}	0.098	C_{l_β}	-3.036

Table 12.6: Major outputs for a converged sample 900 passenger aircraft synthesis

The aircraft's range and endurance performance for varying weights is summarised in table 12.7. Figure 12.18 shows the aircraft's trimmed lift to drag ratio, specific excess power, as well as the trimmed aircraft angle of attack and percent control surface deflection necessary. The maximum elevator deflection is set to 25 degrees. The untrimmed aircraft L/D can be seen in figure 12.19. Both these figures are evaluated at the mid cruise weight and lowest possible static margin. The variation of longitudinal CG position with weight and fuel loading conditions can be seen in figure 12.17.

$\%W_{pld}$	Cruise Range (km)	Loiter Endurance (min)
1.0	14277.14	71.33
0.0	17422.64	418.15
0.0	20577.34	-

Table 12.7: Variation of range and endurance with takeoff weight for a sample 900 passenger long-range aircraft

The aircraft was found to be statically stable when resting on water both at empty and maximum takeoff weight, with a lateral metacentric height of 1.573m and 2.600m respectively. In flight, the aircraft is predicted to be stable in all dynamic modes, which are detailed in table 12.8.

Finally the aircraft was found to have a BFL of 1724m and a landing field length requirement of 1512m. Normal takeoffs require 1694m of waterway. These values are substantially lower than the

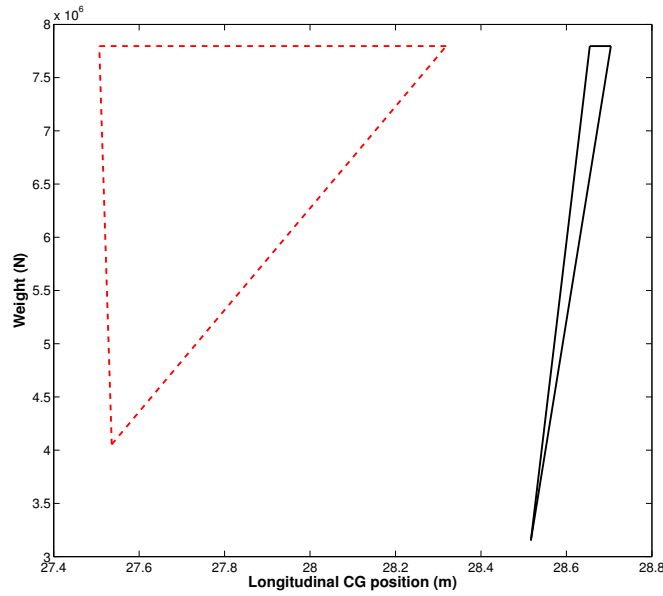


Figure 12.17: CG envelope for a sample 900 passenger, long-range airliner

Mode	Natural Frequency (Hz)	Damping Ratio	Time to Double/Half (s)
Phugoid	0.01377	0.03378	235.76
SPPO	0.63415	0.76256	0.23
Dutch Roll	0.17257	0.07368	8.68
Spiral		Stable	

Table 12.8: Predicted frequencies and damping ratios for the dynamic modes of a 900 passenger sample aircraft

minimum requirement, however they reflect the fact that the engine was sized so that the maximum Mach number could be reached and therefore there was substantially more thrust available at takeoff. The predicted takeoff trim track and the predicted stability limits can be seen in figure 12.20. From these results it is likely that the aircraft may encounter some porpoising when just over the hump speed.

The resulting aircraft can be optimised substantially by increasing the length to beam ratio, allowing both a single floor layout and thus a lower thickness to chord ratio, which would reduce transonic drag and increase its cruise range. The current two-deck design houses a total of 252 passenger on the upper deck and the remaining distributed among 6 cabin bays in the lower deck. The fact that passengers and cargo are placed in the upper deck, away from the cabin can be seen to have an impact on the aircraft's metacentric height, affecting the magnitude of lateral static stability

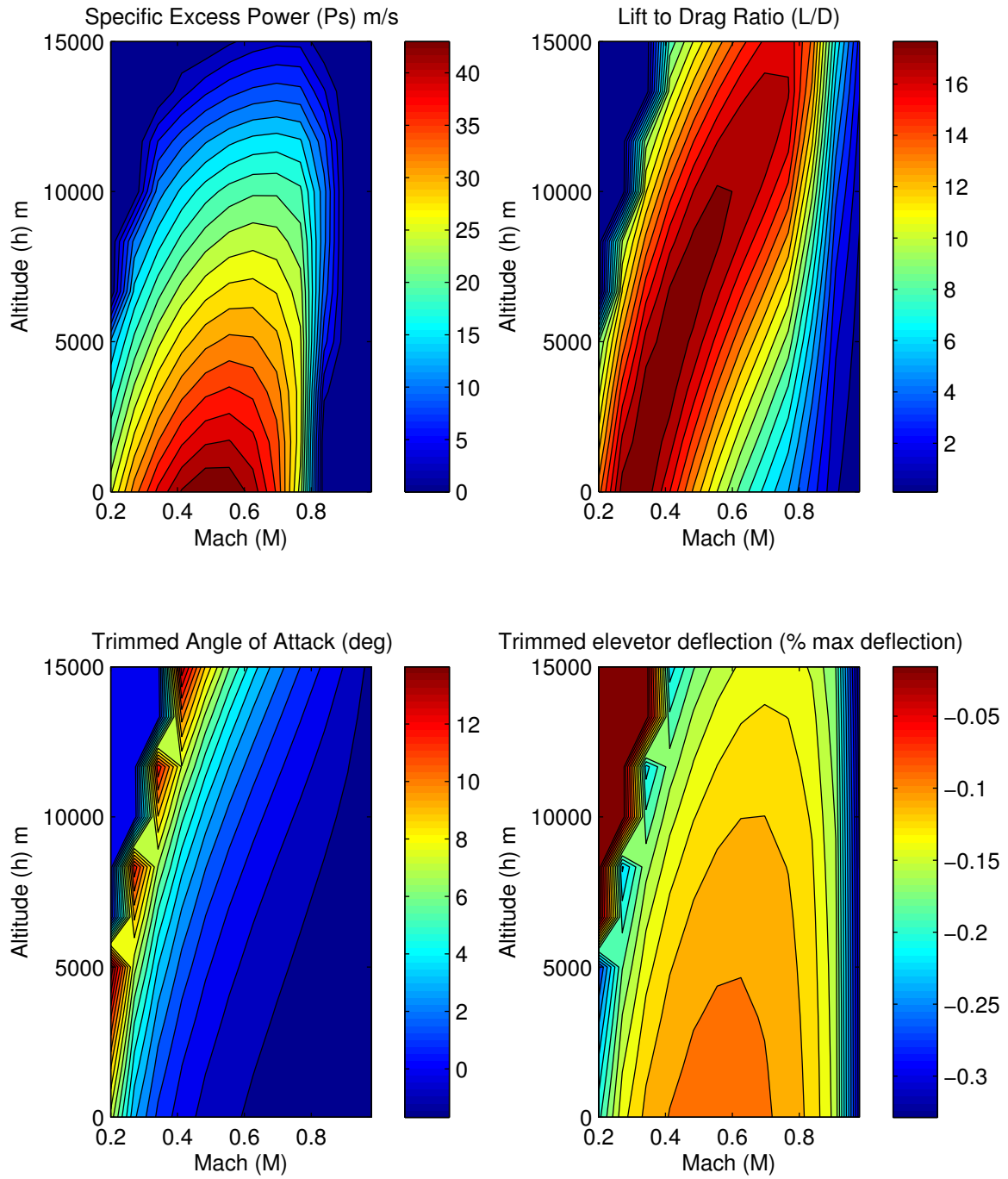


Figure 12.18: Airborne performance & trim characteristics of a 900 passenger sample aircraft at mid cruise

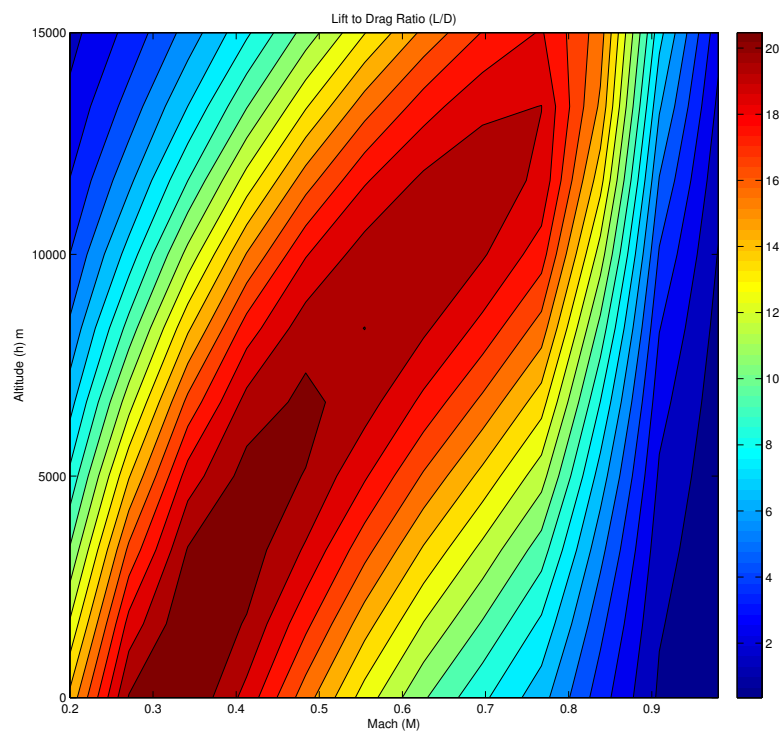


Figure 12.19: Contour plot of the untrimmed lift to drag ratio variation with altitude and cruise mach number for a 900 passenger sample aircraft

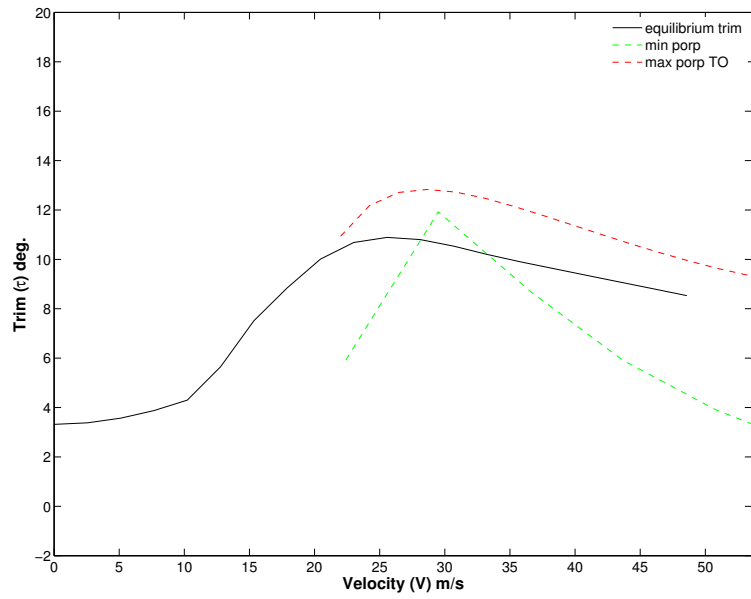


Figure 12.20: Plot of equilibrium trim angle and predicted stability limit variation with velocity during takeoff for a 900 passenger long-range aircraft

while at rest on the water.

Chapter 13

Concluding Remarks

13.1 Discussion

Passenger seaplanes saw extensive use in the pre-1940's era, typically as luxury airliners allowing intercontinental travel. Their popularity has since declined and the vast majority of aircraft now operate from land. This can be attributed to the large performance and weight penalties their unique design exhibited in the past. Waterborne aircraft are now only used for niche applications and few attempts have been made to improve their performance using modern technology.

The preceding chapters presented the methods necessary for a modern seaplane to be designed and a novel design concept aimed at designing a waterborne aircraft that could possibly compete with or outperform the current generation of airliners. Numerous designs were generated using the synthesis code described in chapter 11, two of which can be seen as sample aircraft in chapter 12.

Based on these designs, the general performance characteristics of the proposed aircraft can be deduced. The aircraft's cruise performance, as given by its lift to drag ratio, is optimum at Mach numbers between 0.65 - 0.75 at a typical cruise altitude of 35,000 ft, while current state of the art passenger aircraft typically operate at an optimum Mach number of 0.8 - 0.85 at these altitudes. The optimum cruise Mach number does however increase with altitude. The maximum untrimmed L/D is around 19 - 21, a value comparable to that expected of proposed future blended wing body airliners.

This reduction in optimum cruise Mach number is most likely a side effect of the need to use lower wing loadings (of the order of $5000 - 6000 N/m^2$) so that approach speed constraints can be met and to compensate for the lack of landing flaps, which cannot be used on an aircraft without a horizontal

stabiliser. The choice to blend the hull with the rest of the aircraft and to fair the step can however be deemed largely successful, substantially reducing viscous and interference drag during flight.

Moreover the aircraft's lengthwise cross sectional area distribution is consistently smooth, leading to reasonable wave drag rise magnitude at transonic speeds. The choice of hull length to beam ratio however was found to have a considerable effect on the aircraft's cross-sectional thickness and thus the expected drag divergence Mach number. Increasing the hull's L/B often led to a decrease in thickness to chord ratio and thus a higher drag divergence Mach number, however the associated increased length also increased the aircraft wetted area and thus its viscous drag, in addition to affecting the aircraft's MTOW. Ultimately the optimum L/B necessary to balance hydrodynamic, as well as high speed and low speed aerodynamic concerns was found to largely depend on the number of passengers required and the packaging constraints imposed.

The need to trim the aircraft during flight does however have a substantial impact on the cruise L/D , as trim drag is rather severe. This is a consequence of having to place the propulsive units some distance above the aircraft body, producing a nose down pitching moment. Counteracting this moment necessitates that the aircraft's elevators and elevons, which cover the entire outer wing, be deflected by about -2.5 to -4.0 degrees. The resulting loss of lift and increase in trim drag can often lead to a reduction of maximum lift to drag ratio to values in the range of 17 - 15. Moreover, as the centre of gravity of the fuel system is somewhat aft of the payload and empty aircraft centres of gravity, the aircraft's static margin becomes larger as fuel is consumed during flight, further reducing the aircraft's efficiency. This entire problem could be somewhat mitigated in future designs by allowing the aircraft to remain statically unstable during flight, and using fly by wire controls to automatically stabilise the aircraft when perturbed.

The aerodynamic performance of the centre section was found to rely heavily on the hull design chosen. Use of large afterbody keel angles resulted in the centre section airfoils having a negative camber distribution. This could often be remedied by simply increasing the ratio $|Z'_u/Z'_l|$ defined for the hull airfoils. Such a move does however often result in a more aft loaded airfoil, further increasing the aircraft's nose down moment and thus trim drag. The need to warp the hull forebody for hydrodynamic reasons also affects the hull section airfoils, imparting increasingly negative camber distributions forward of the step. This effect became less apparent when using high length to beam ratio hulls and high centre section leading edge sweep angles. Due to this inherent inflexibility in designing the airfoils and the impossibility of setting the remaining wing section at a negative angle to the hull, due to both packaging and takeoff performance constraints, the hull section was often found to produce substantially lower amounts of lift than the remaining aircraft, not only affecting

the aircraft's lift performance but also the planform's efficiency.

The methods used for the aircraft synthesis are considered to be mostly adequate, providing an accurate initial estimate for the aircraft's characteristics. The only methodology severely lacking much needed accuracy, despite extensive efforts to find or generate one that would be more appropriate, is the centerbody weight estimation method used.

The empirical weight estimation methodology used, is based on a structural concept which assumes that composite panels on the top and bottom of a BWB centre body bear both shear loads and pressure loads. The structural design utilised for the proposed aircraft assumes that pressure loads are carried solely by multiple internal conjoined pressure vessels, while shear loads are carried by a thin composite or metallic skin, a configuration which was shown by Mukhopadhyay [60] to further reduce structural weight.

A weight penalty equal to 60% of a typical fuselage's structural weight was subsequently applied, as suggested by Raymer [67]. This approach however does not account for the effect that the hull shape and dimensions have on the hydrodynamic impact loads and thus the hull weight penalty. In past designs, such as the Canadair CL-215 [68], the deadrise angle has been chosen such that an optimum balance between structural weight and aerodynamic performance was achieved, something which is not possible given the lack of weight dependence on deadrise angle, beam loading and landing approach velocity. The effect of hull length to beam ratio on structural weight, as found by Benschoter [9], is modelled to a certain extent by the fuselage weight estimation relation used, however the exact order of the relation is by no means replicated. Neither is the closely related effect of increasing the beam loading and thus reducing the maximum impact loading. The weight of the step fairing was also not considered, assuming that the hull weight penalty was sufficient.

Overall the aircraft empty weight was found to be higher than that of even previous generation airliners intended for similar missions. This conclusion however is by no means definitive as a good estimate of the hull empty weight could not be obtained. What is certain however is that the resulting aircraft empty weight is substantially lower than the trend of empty weight vs maximum takeoff weight as defined by existing seaplane designs, indicating a clear improvement over past seaplane designs.

Using a fully parametrized design, the aircraft shape can be quickly and easily altered, an ability that will prove invaluable in future optimisation attempts. Moreover a conscious decision to limit the number of parameters to the extent possible makes the effects of changing a single parameter instantly and easily visible. Although the resulting code itself lacks any graphical capability, outputs can be

easily displayed using alternate programs such as MATLAB.

The proposed design also proved to have substantial amounts of empty volume throughout the airframe, allowing for an extremely flexible packaging of the aircraft. The need to raise the wing outboard of the passenger cabin also provided an elegant solution to the emergency egress issue that is very prominent in existing flying wing and BWB designs. The aircraft's ability to operate from water makes ditching considerations obsolete and in case of a crash landing the strengthened hull may actually improve the level of viability.

The majority of aerodynamic calculations were performed using the Vortex Lattice Method. Although this method has proven extremely accurate for arbitrary planforms consisting of thin airfoils, its applicability to the inner part of the wing is questionable particularly for designs using lower hull length to beam ratios which often feature very thick centre body airfoil sections ($(t/c)_o > 0.15$). This particular method however still remains the best choice over three dimensional panel codes due to the associated saving in computational time. Moreover although a rather limited number of panels was used, in order to cut down on running time, the incremental increase in accuracy achieved by doubling the number of panels was found to be insignificant.

The Prandtl - Glauert correction is used to model the effect of compressibility throughout, which for Mach numbers higher than 0.75 may lead to an over prediction of pressure forces. Using alternate theoretical or empirical compressibility corrections could improve accuracy, however such equations were not used as their accuracy for the planform in question could not be verified.

The VLM code is also used to provide the aircraft's longitudinal and lateral stability derivatives. The longitudinal derivatives obtained can be assumed to be rather accurate, as the lifting surface modelled bears close resemblance to the intended design. That cannot however be said of the lateral derivatives, where the effects of body thickness, the engine pylons and the engine pods themselves are not modelled. The vertical profile of a body is normally inserted as a flat surface along the aircraft's centreline to model the effects of body thickness on the aircraft's response to a perturbation in sideslip, something that was avoided due to the rather severe computational cost associated with such a move. Instead the vertical projection of the wing surface itself, which in certain sections features rather heavy dihedral, was considered to be sufficient. The engine pylons, which when placed aft of the CG act as additional stabilising fins, were also omitted thus leading to an under prediction of the aircraft's lateral stability and roll-yaw coupling.

The aircraft's viscous and wave drag were estimated using methodologies that have proven to be very accurate in the past. The novel method presented for estimating the drag penalty of using a

stepped airfoil allows for the simple and rapid estimation of the aircraft's aerodynamic characteristics when operating with an unfaired step. The expression was however based on the comparison between CFD results for stepped airfoils and the drag of these test airfoils when featuring a 9:1 step fairing, as predicted using a panel code coupled with an iterative viscous-inviscid matching algorithm. The use of CFD results for the drag of the faired airfoils would have been a safer choice, however there was insufficient time to complete a large number of additional CFD computations. Moreover, lacking experimental data, the CFD results themselves could not be validated but were considered accurate based on previous experience with the specific problem formulation. The resulting methodology may therefore suffer from inaccuracies stemming from the mis-estimation of the faired step drag, however the percentage increase in sectional drag does lie within the bounds presented by the available literature.

Both the VLM and drag prediction methods are also likely to mis-predict the aircraft's aerodynamic characteristics at high angles of attack, as trailing edge separation and stall cannot be modelled. Furthermore, minor errors may stem from the use of the empirical DATCOM methods, to predict the effect of leading and trailing edge devices, which are strictly only applicable to straight tapered wings. The VLM could be used instead but at a substantial computational cost and uncertain levels of accuracy.

The aircraft's takeoff and landing length were identified as major sizing constraints early on. A set of simple equations were therefore generated based on experimental towing tank data, in order to predict the thrust to weight ratio necessary for a predefined takeoff distance to be met at the initial sizing phase. The balanced field length was chosen to comply with current airworthiness directives, despite the fact that only normal takeoff distances are quoted in current seaplane performance sheets. The takeoff constraint was seldom the defining thrust constraint for field lengths larger than $3.0km$, particularly when the aircraft had to be capable of reaching high transonic maximum Mach numbers.

A number of relations were also generated to allow the rapid hydrodynamic evaluation of an arbitrary hull's resistance and pitching moment characteristics. These were used in conjunction with the aerodynamic data available to predict the aircraft's running attitude and resistance variation with velocity during takeoff and landing manoeuvres. The resistance equations showed a good agreement with the source data both for high and low speeds, however the resulting values were found to diverge for the very low beam loadings typically encountered just prior to takeoff at high speed.

The resistance values obtained are heavily dependent upon the hull's predicted running trim. Two

distinct methods were generated to predict the equilibrium trim angle, one based on the centre of pressure location and one on the trim necessary to produce a given pitching moment. As shown on section 8.4 the centre of pressure relation was found to be most accurate in both the displacement and planing regimes. The level of accuracy possible using these methodologies remains questionable, however both the methods presented can give the user a good idea of the hull's performance on water during takeoff and landing and can most certainly be used to ensure that the lower porpoising stability trim limit is not exceeded. The centre of pressure method was particularly accurate in predicting the hump speed, the velocity where the aircraft starts riding on the step, however the predicted trim angles leave much to be desired.

The resulting takeoff balanced field length and landing distances were often found to be substantially lower than the predefined length, partly due to the use of thrust reversal and spoilers, the effects of which were not considered when generating the initial sizing methodology. The initial assumption that the landing length would almost always be lower than the equivalent BFL was verified, particularly when thrust reversal and spoilers were employed. The predicted resistance values however will almost certainly be somewhat lower than in reality, leading to a rather certain underestimation of the takeoff distance and overestimation of landing distance requirements, which however cannot be quantified.

In predicting the hull's hydrodynamic stability, three distinct equations were produced to characterise the upper and lower porpoising stability limits. The validity of these equations for a broader range of hull shapes remains to be seen as the limited number of data points and large number of parameters necessary to generate a relatively accurate regression may have led to substantial over fitting and loss of generality. This is particularly the case for the two upper trim limit equations, especially the decreasing trim one. Therefore extreme care should be taken when considering the resulting trim limits.

Overall the methods used can give the designer a decent initial view of the hydrodynamic performance of the aircraft during takeoff and landing and considering the effect of varying aerodynamic configurations. They do not however yet present an accurate enough alternative to tank testing or the use of a hull shape whose performance characteristics are known.

Finally based on the general performance characteristics of the proposed design, one can see that this type of aircraft may be most suitable for cargo transport and air mobility operations. One extremely viable application would be as a future C-5 Galaxy replacement capable of operating within

the Sea-Base framework. Instead, use of such seaplanes as commercial airliners may be hindered by the lower optimum cruise Mach numbers and increased structural weight which would impede its financial viability.

13.2 Conclusions

The seaplane designed for this research program is a novel concept for a waterborne aircraft that could possibly compete or outperform the current generation of aircraft. An extensive feasibility study was performed, identifying the major advances in technology that could improve the performance of a modern passenger seaplane, making it competitive. A brief review of other applications for such an aircraft was conducted and the major operational issues and constraints that a modern passenger seaplane would be faced with were identified, along with some preliminary proposals of how these issues could be faced.

The resulting aircraft configuration is a flying wing where the hull is seamlessly blended into the airfoil shaped cross-sections of the aircraft. The entire aircraft shape is defined in terms of a limited number of control parameters, allowing the rapid creation of multiple derivative aircraft. loosely based on the predefined baseline configuration. The shape was generated by blending varying parameterised airfoil sections to produce a predefined wing planform. The aircraft's packaging requirements are incorporated into the shape generation methodology, ensuring that there is sufficient volume to accommodate the predefined number of passenger and cargo. The synthesis subsequently allowed great flexibility in automatically locating all other systems wherever sufficient free volume is found.

A novel method for estimating the field length necessary for seaplane to takeoff from water was generated using the vast amount of hull towing tank test data available. It relates Balanced Field Length to a number of initial aircraft design parameters, allowing for a takeoff constraint to be included in the carpet plot when initially sizing the aircraft. This same methodology can also be used to evaluate the takeoff performance of the aircraft in the preliminary design phase, however a tendency to over-predict takeoff distances has been identified.

By not using tip floats or the large stabiliser surfaces, typically present on conventional seaplane designs, the aircraft weight could be reduced despite the persistent weight penalty associated with the strengthening of the hull structure to withstand landing impact loads. The hull weight estimation

methodology suggested provides a rather crude approximation of what the weight found in the detailed design phase might be.

The aerodynamic characteristics of the aircraft were identified using a VLM code, allowing the accurate prediction of pressure loads both in and out of ground effect. A novel methodology to estimate the aerodynamic effects of incorporating a step on the hull's airfoil sections was generated using the results of CFD computations performed on airfoil sections of varying shape at different angles of attack as well as Mach and Reynolds numbers.

A number of novel empirical methodologies for predicting the hydrodynamic behaviour of a parametric hull were also generated based on experiment data. These equations were generated using a stepwise multivariate regression algorithm, and are capable of predicting to varying degrees of accuracy the hull's resistance, running trim angle and the trim angles at which the onset of porpoising is likely to occur. These methods were subsequently used to provide a more accurate prediction of the resulting aircraft design's takeoff and landing distance requirements. This work further highlights areas of research in hydrodynamics, which are worthwhile of further significant research effort.

The resulting aircraft designs were found to vastly outperform most seaplanes currently available on the market, exhibiting aerodynamic performance characteristics more akin to the current generation of airliners or even proposed future designs. They were however also found to suffer from a substantial trim drag increment necessary for steady level flight which somewhat reduced their aerodynamic efficiency.

The wing loading values that could be achieved using the proposed design in order for takeoff and landing constraints to be met, were substantially lower than those of most current generation airliners. Consequently there was a substantial reduction of the Mach number at which cruise lift to drag ratio is maximised. Higher wing loadings could however be achieved if horizontal stabilisers or a V-tail are utilised in future iterations, allowing for high lift devices to be placed along parts of the wing trailing edge.

Finally the increased weight necessary for waterborne operations, although somewhat minimised, remains a major issue. Overall, the adoption of seaplanes for airline operations as we know them today seems doubtful, however there may be a future for this concept in the cargo transport and strategic airlift sectors.

13.3 Further Work

A number of research issues and areas could not be addressed in this research program due either the lack of time or resources. A major aspect of the original research plan that could not be completed due to time constraints is design of an optimised aircraft using the synthesis algorithm that was developed. The multi-point optimiser packaged within MATLAB can be used as the synthesis code written is already capable of operating within it.

The time necessary for performing a single synthesis is rather substantial, ranging anywhere from 5 to 20+ minutes. By parallelising portions of the code, particularly the VLM implementation, the running time could be cut down substantially when used on most multicore systems, therefore yielding an optimised result in a few days rather than months.

The weight estimation methodology used in the present synthesis is the one aspect severely lacking accuracy. As the impact loads likely to be experienced during landings can be predicted using either the airworthiness requirements or Wagner's added mass formulation, Finite Element analyses can be performed to calculate the stress distribution along the hull. The weight of the entire centre section can thus be more accurately approximated. Repeating this analysis for numerous different hull shapes, beam loading coefficients and approach speeds, a better empirical methodology could be also generated.

Further work must also be done in the field of hydrodynamic prediction. The added-mass strip theory formulation presented previously could possibly produce more accurate results than the proposed empirical relations, if an empirical relation allowed the accurate prediction of the wake's shape aft of a step where the water flow has separated. If the dynamic behaviour of the wake could also be easily characterised, the porpoising instability onset for seaplane hulls could be more accurately predicted computationally.

The handling qualities of the designed aircraft could not be characterised in time. By designing a model of the aircraft for use in simulator software such as X-Plane, a full set of flight tests could be performed. The accuracy of X-Plane in modelling the waterborne behaviour of aircraft should also be investigated and if it is deemed accurate, sea trials should also be performed, evaluating the handling characteristics on water and under varying sea states.

Given that the aircraft seemed to perform optimally at $M \approx 0.7$, the use of prop fans should be

considered as they have been found to outperform high bypass ratio engines at such cruise speeds. Finally a full cost analysis of the aircraft should be done, in hopes of reinforcing the conclusions reached both in the feasibility study and by Denz et al. [25].

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Appendix A

Seaplane Database

Model ID	Source Report	Model ID	Source Report
11	NACA TN-464	11-A	NACA TN-470
11-B	NACA TN-545	11-C	NACA TN-535, TN-538
11-C-7	NACA TN-541	11-C-8	NACA TN-541
11-C-9	NACA TN-541	11-C-10	NACA TN-541
11-C-11	NACA TN-535	11-C-12	NACA TN-535
11-C-13	NACA TN-535	11-C-45P	NACA TN-538
11-C-30-ST	NACA TN-538	11-G	NACA TN-531
12	NACA TN-291	13	NACA TN-491
14	NACA TN-291	15	NACA TN-491
16	NACA TN-471	18	NACA TN-681, TN-594
22	NACA TN-488	22-A	NACA TN-504
26	NACA TN-512	35	NACA TN-504
35-A	NACA TN-551	35-B	NACA TN-551
36	NACA TN-638	40-AC	NACA Re-543
40-AD	NACA Re-543	40-AE	NACA Re-543
40-BC	NACA Re-543	40-BE	NACA Re-543
41-A	NACA TN-563	41-D	NACA TN-656
44	NACA TN-566	47	NACA TN-590
52	NACA TN-576	57-A	NACA TN-716
57-B	NACA TN-716	57-B-5	NACA TN-716
61-A	NACA TN-656	62-D	NACA TN-725

Model ID	Source Report	Model ID	Source Report
62-E	NACA TN-725	62-F	NACA TN-725
66	NACA TN-858	73	NACA TN-656
75	NACA TN-668	83	NACA TN-836
84-EF-3	NACA ARR-3I15	126A-1	NACA ARR-3B23
126A-2	NACA ARR-3B23	126A-3	NACA ARR-3B23
126B-1	NACA ARR-3B23	126B-2	NACA ARR-3B23
126C-1	NACA ARR-3B23	126C-2	NACA ARR-3B23
126C-3	NACA ARR-3B23	144	NACA ARR-3J23
145	NACA ARR-3J23	146	NACA ARR-3J23
184	NACA TN-1182	185	NACA ARR-L5G19
185-A	NACA TN-1182	L/B = 5.5	NACA TN-1182
L/B = 7.0	NACA TN-1182	L/B = 8.5	NACA TN-1182
L/B = 10.5	NACA TN-1182	Shetland	NACA TN-1182
294-79	NACA TN-1182	339-1	NACA TN-1182
339-15	NACA TN-1182	339-20	NACA TN-1182
339-22	NACA TN-1182	339-23	NACA TN-1182
339-46	NACA TN-1182	406	NACA TN-1182

Appendix B

Modified Airfoil Curves

If the original airfoil top curve is given by $z(x)$ and the packaging constraint for the final curves ($z'(x)$) is such that $z'(x) \geq z_c(x)$ then $|x - X_u|(z_c(x) - z(x))$ is minimised at $x_f < X_u$ and $x_a > X_u$.

If $z_c(x_f) > z(x_f)$ and $z_c(x_a) > z(x_a)$:

$$z'(x) = \begin{cases} \sum_{i=1}^3 a_i x^{(2i-1)/2} & \text{if } 0 \leq x \leq x_f \\ \sum_{i=1}^4 b_i x^{(2i-1)/2} & \text{if } x_f \leq x \leq x_a \\ \sum_{i=1}^4 c_i x^{(2i-1)/2} & \text{if } x_a \leq x \leq 1 \end{cases} \quad (\text{B.1})$$

where the constants a_i, b_i and c_i are obtained by solving the following systems of equations

$$\begin{aligned} \sum_{i=1}^4 b_i x_f^{(i-0.5)} &= z_c(x_f) \\ \sum_{i=1}^4 b_i x_a^{(i-0.5)} &= z_c(x_a) \\ \sum_{i=1}^4 b_i X_u^{(i-0.5)} &= \max(Z_u, z_c(x_f), z_c(x_a)) \\ \sum_{i=1}^4 b_i (i - 0.5) X_u^{(i-1.5)} &= 0 \end{aligned} \quad (\text{B.2})$$

$$\begin{aligned} a_1 &= \sqrt{2R_u} \\ \sum_{i=1}^3 a_i \sqrt{x_f}^{(2i-1)} &= z_c(x_f) \\ \sum_{i=1}^3 a_i (2i - 1) \sqrt{x_f}^{(2i-2)} &= 2\sqrt{x_f} \sum_{j=1}^4 b_j (j - 0.5) x_f^{(j-1.5)} \end{aligned} \quad (\text{B.3})$$

$$\begin{aligned}
\sum_{i=1}^4 c_i &= Z_{te} + 0.5\Delta Z_{te} \\
\sum_{i=1}^4 c_i x_a^{(i-0.5)} &= z_c(x_a) \\
\sum_{i=1}^4 c_i(2i-1) &= -2 \tan(\beta_{te}/2 + \alpha_{te}) \\
\sum_{i=1}^4 c_i(i-0.5)x_a^{(i-1.5)} &= \sum_{j=1}^4 b_j(j-0.5)x_a^{(j-1.5)}
\end{aligned} \tag{B.4}$$

If $z_c(x_f) > z(x_f)$ and $z_c(x_a) < z(x_a)$:

$$z'(x) = \begin{cases} \sum_{i=1}^3 a_i x^{(2i-1)/2} & \text{if } 0 \leq x \leq x_f \\ \sum_{i=1}^3 b_i x^{(2i-1)/2} & \text{if } x_f \leq x \leq x_a \\ z(x) & \text{if } x_a \leq x \leq 1 \end{cases} \tag{B.5}$$

where the constants a_i and b_i are obtained by solving the following systems of equations

$$\begin{aligned}
\sum_{i=1}^3 b_i x_f^{(i-0.5)} &= z_c(x_f) \\
\sum_{i=1}^3 b_i X_u^{(i-0.5)} &= \max(Z_u, z_c(x_f)) \\
\sum_{i=1}^3 b_i(i-0.5)X_u^{(i-1.5)} &= 0
\end{aligned} \tag{B.6}$$

$$\begin{aligned}
a_1 &= \sqrt{2R_u} \\
\sum_{i=1}^3 a_i \sqrt{x_f}^{(2i-1)} &= z_c(x_f) \\
\sum_{i=1}^3 a_i(2i-1)\sqrt{x_f}^{(2i-2)} &= 2\sqrt{x_f} \sum_{j=1}^3 b_j(j-0.5)x_f^{(j-1.5)}
\end{aligned} \tag{B.7}$$

If $z_c(x_f) < z(x_f)$ and $z_c(x_a) > z(x_a)$:

$$z'(x) = \begin{cases} z(x) & \text{if } 0 \leq x \leq x_f \\ \sum_{i=1}^3 b_i x^{(2i-1)/2} & \text{if } x_f \leq x \leq x_a \\ \sum_{i=1}^4 c_i x^{(2i-1)/2} & \text{if } x_a \leq x \leq 1 \end{cases} \tag{B.8}$$

where the constants b_i and c_i are obtained by solving the following systems of equations

$$\begin{aligned}
\sum_{i=1}^3 b_i x_a^{(i-0.5)} &= z_c(x_a) \\
\sum_{i=1}^3 b_i X_u^{(i-0.5)} &= \max(Z_u, z_c(x_f), z_c(x_a)) \\
\sum_{i=1}^3 b_i (i-0.5) X_u^{(i-1.5)} &= 0
\end{aligned} \tag{B.9}$$

$$\begin{aligned}
\sum_{i=1}^4 c_i &= Z_{te} + 0.5 \Delta Z_{te} \\
\sum_{i=1}^4 c_i x_a^{(i-0.5)} &= z_c(x_a) \\
\sum_{i=1}^4 c_i (2i-1) &= -2 \tan(\beta_{te}/2 + \alpha_{te}) \\
\sum_{i=1}^4 c_i (i-0.5) x_a^{(i-1.5)} &= \sum_{j=1}^3 b_j (j-0.5) x_a^{(j-1.5)}
\end{aligned} \tag{B.10}$$

Appendix C

Stepped Airfoils

ID No.	t/c	h_s/c	$\delta_k(deg.)$	$\tau_o(deg.)$	M_∞	Re
1	0.1625	0.0085	5.9673	0.3376	0.4663	31.9077×10^7
2	0.1499	0.0074	7.6964	0.7948	0.6959	47.6963×10^7
3	0.1722	0.0172	6.1607	0.4130	0.5420	37.0768×10^7
4	0.1703	0.0256	4.5434	0.3150	0.4043	27.6860×10^7
5	0.1196	0.0288	6.5871	0.8141	0.3198	21.8971×10^7
6	0.1720	0.0152	6.3954	0.8317	0.6019	41.2157×10^7
7	0.1699	0.0296	5.5703	0.1451	0.3233	22.1368×10^7
9	0.1703	0.0214	6.1719	0.9762	0.2392	16.3777×10^7
10	0.1753	0.0278	5.9192	0.4092	0.6221	42.5972×10^7
11	0.1903	0.0248	8.4034	0.4239	0.5351	36.6377×10^7
12	0.1745	0.0210	6.5456	0.5701	0.5482	12.5115×10^7
13	0.1074	0.0127	4.1936	0.1943	0.4362	9.9550×10^7
14	0.1266	0.0074	6.5403	0.6014	0.4128	9.4222×10^7
15	0.1042	0.0060	7.6008	0.2754	0.5566	12.7033×10^7
16	0.1117	0.0200	8.0632	1.1817	0.3411	7.7846×10^7
17	0.1881	0.0271	8.5744	1.0117	0.2291	5.2293×10^7
19	0.1730	0.0290	5.4392	0.7991	0.2671	6.0973×10^7

Table C.1: Basic design parameters for test airfoils 1-19

ID No.	t/c	h_s/c	$\delta_k(deg.)$	$\tau_o(deg.)$	M_∞	Re
20	0.1803	0.0201	4.2603	0.8468	0.3065	6.9958×10^7
21	0.1453	0.0280	4.4866	0.6949	0.5784	13.2017×10^7
22	0.1324	0.0115	5.2592	0.7690	0.6484	14.7993×10^7
23	0.1656	0.0068	5.3019	0.4011	0.2959	40.5160×10^7
24	0.1830	0.0102	8.2292	0.1513	0.3059	41.8907×10^7
25	0.1834	0.0205	5.1250	0.9047	0.5975	81.8277×10^7
26	0.1394	0.0125	8.8549	1.3220	0.6434	88.1087×10^7
27	0.1104	0.0198	6.8192	0.4476	0.5135	70.3205×10^7
28	0.1475	0.0257	6.0364	0.6959	0.3295	45.1236×10^7
29	0.1043	0.0249	4.9009	0.5286	0.4323	59.1986×10^7
31	0.1767	0.0287	6.2326	0.1392	0.5038	1.1500×10^7
32	0.1070	0.0297	7.4277	1.1672	0.6086	1.3891×10^7
33	0.1293	0.0070	5.5080	0.0467	0.4351	0.9930×10^7
34	0.1511	0.0137	7.8175	1.2489	0.3364	0.7679×10^7
35	0.1787	0.0151	5.7150	0.9851	0.5220	1.1914×10^7

Table C.2: Basic design parameters for test airfoils 20-35

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
1	-0.4664	-0.13619	0.1753	0.3646	0.58018
2	-0.56936	-0.46065	-	-	0.29682
3	0.10627	0.31778	0.59599	0.78981	0.87785
4	0.055383	0.33388	0.62478	0.82044	1.11000
5	-0.32434	-0.10239	0.17112	0.37265	0.67929
6	-0.50036	-0.32984	0.005248	0.23721	0.47099
7	-0.31242	-0.090253	0.1311	0.26394	0.46753
9	-0.63699	-0.35808	-0.050963	0.15976	0.47506
10	0.20595	0.49663	0.57429	0.64717	0.63582
11	-	0.44587	0.64889	0.70928	0.80553
12	0.15862	0.30666	0.49064	0.70269	0.83062
13	-0.4291	-0.079429	0.27459	0.50945	0.85111
14	-0.4118	-0.063099	0.28778	0.5192	0.85284
15	-0.43665	-0.17768	0.15899	0.39324	0.68986
16	-0.30359	-0.059521	0.19201	0.38411	0.68255
17	-	-	0.61651	0.71392	0.87771
19	0.10725	0.35169	0.57177	0.75199	1.0165

Table C.3: Lift coefficient (C_l) results from CFD analysis for test airfoils 1-19

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
20	0.11770	0.40570	0.68829	0.87610	1.14370
21	-0.07225	0.17428	0.48630	0.70669	0.87131
22	-0.23021	-0.08598	0.05002	0.25809	0.48221
23	-0.50915	-0.17239	0.16683	0.39024	0.71164
24	-0.08747	0.20058	0.49218	0.69362	0.97173
25	0.37742	0.53929	0.64221	0.75335	0.76162
26	-0.19764	0.00925	0.35048	0.52250	0.66274
27	-0.54282	-0.24004	0.11411	0.35750	0.72254
28	-0.03753	0.21485	0.47784	0.66459	0.95458
29	-0.33046	-0.02742	0.28821	0.50408	0.82655
31	0.12623	0.23206	0.36389	0.50729	0.77284
32	-	-0.17437	0.00029	0.17647	0.49398
33	-0.28337	0.03666	0.35937	0.56691	0.85513
34	-0.35358	-0.07293	0.22551	0.42956	0.74227
35	-	-	0.07159	0.40592	0.61754

Table C.4: Lift coefficient (C_l) results from CFD analysis for test airfoils 20-35

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
1	0.01747	0.01448	0.01308	0.01327	0.01454
2	0.05537	0.02118	-	-	0.04093
3	0.01801	0.01567	0.01585	0.01776	0.03440
4	0.01834	0.01586	0.01550	0.01628	0.01912
5	0.02803	0.02108	0.01735	0.01607	0.01565
6	0.04260	0.02424	0.01911	0.01758	0.02608
7	0.02895	0.02232	0.01889	0.01771	0.01691
9	0.02329	0.01860	0.01581	0.01473	0.01423
10	0.02290	0.02194	0.02318	0.03142	0.05671
11	-	0.02608	0.02434	0.02347	0.02845
12	0.02259	0.02009	0.01924	0.02017	0.03476
13	0.01243	0.01013	0.00971	0.01042	0.01318
14	0.01258	0.01031	0.00985	0.01055	0.01331
15	0.01427	0.01075	0.00976	0.01035	0.01701
16	0.02407	0.01931	0.01622	0.01536	0.01562
17	-	-	0.02298	0.02226	0.02203
19	0.02104	0.01905	0.01797	0.01805	0.01990

Table C.5: Drag coefficient (C_d) results from CFD analysis for test airfoils 1-19

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
20	0.01585	0.01475	0.01525	0.01646	0.01986
21	0.02376	0.01878	0.01737	0.01853	0.03611
22	0.04207	0.02056	0.01318	0.01305	0.03010
23	0.01093	0.00913	0.00857	0.00891	0.01057
24	0.01423	0.01236	0.01212	0.01268	0.01462
25	0.02441	0.01882	0.01852	0.02674	0.05208
26	0.03453	0.01993	0.01620	0.02038	0.04373
27	0.02624	0.01841	0.01460	0.01366	0.01455
28	0.02088	0.01729	0.01556	0.01540	0.01661
29	0.01928	0.01503	0.01342	0.01335	0.01490
31	0.03661	0.02596	0.02495	0.02496	0.02900
32	-	0.03193	0.02598	0.02332	0.02444
33	0.01299	0.01160	0.01217	0.01370	0.01871
34	0.02266	0.01815	0.01636	0.01628	0.01793
35	-	-	0.01923	0.01884	0.02183

Table C.6: Drag coefficient (C_d) results from CFD analysis for test airfoils 20-35

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
1	0.06367	0.05848	0.04735	0.03405	-0.00322
2	0.04433	0.02229	-	-	-0.09081
3	0.13607	0.10569	0.08232	0.06453	0.00568
4	0.12229	0.11256	0.10019	0.09051	0.07316
5	0.09673	0.08584	0.07751	0.07318	0.06523
6	0.06453	0.03488	0.01971	0.00905	-0.03515
7	0.09314	0.08124	0.06010	0.04012	0.00959
9	-0.00175	-0.00981	-0.01589	-0.01925	-0.02461
10	0.16251	0.13843	0.06870	0.02798	-0.02541
11	-	0.14692	0.11426	0.07700	0.02127
12	0.13962	0.09886	0.06125	0.04975	-0.00041
13	0.05183	0.04748	0.04398	0.03079	0.03181
14	0.06899	0.06682	0.06410	0.06081	0.05088
15	0.09433	0.06369	0.05016	0.04213	0.01355
16	0.07879	0.06803	0.05471	0.04833	0.03867
17	-	-	0.06303	0.04222	0.01428
19	0.09411	0.08272	0.06436	0.05724	0.04397

Table C.7: Moment coefficient (C_m) results from CFD analysis for test airfoils 1-19

ID No.	Angle of Attack				
	$\alpha = -5$ deg.	$\alpha = -2$ deg.	$\alpha = 1$ deg.	$\alpha = 3$ deg.	$\alpha = 6$ deg.
20	0.12193	0.01124	0.10005	0.09134	0.07586
21	0.14610	0.12395	0.10650	0.09258	0.04349
22	0.12289	0.09531	0.03541	0.01293	-0.03422
23	-0.00685	-0.00740	-0.00825	-0.00986	-0.01574
24	0.10077	0.09177	0.08075	0.07202	0.05880
25	0.18739	0.14641	0.08485	0.04987	-0.00610
26	0.11031	0.08938	0.07505	0.05225	-0.00691
27	0.07142	0.06604	0.06429	0.06216	0.05397
28	0.10146	0.09325	0.08100	0.07278	0.06030
29	0.10346	0.09738	0.08921	0.08345	0.07167
31	0.14404	0.06394	0.02278	0.00549	-0.01452
32	-	0.07253	0.03747	0.02420	0.00547
33	0.06344	0.05547	0.04708	0.03909	0.02158
34	0.07680	0.06849	0.05818	0.05308	0.04298
35	-	-	-0.04117	-0.04504	-0.06192

Table C.8: Moment coefficient (C_m) results from CFD analysis for test airfoils 20-35

Appendix D

Stepwise Multiple Regression

Stepwise regression works by successively adding regressors (x_i) to a multiple least squares regression model, giving:

$$\hat{y} = a_0 + a_1x_1 + a_2x_2 + a_3x_3 + \dots + a_kx_k \quad (\text{D.1})$$

The statistical significance of adding a parameter x^* to the regression model is given by the partial f statistic.

$$f(x^*|x_1, x_2, \dots, x_k) = (n - k - 2) \frac{\text{SSR}(x_1, x_2, \dots, x_k, x^*) - \text{SSR}(x_1, x_2, \dots, x_k)}{\text{SSE}(x_1, x_2, \dots, x_k, x^*)} \quad (\text{D.2})$$

where n is the number of observations, $k + 2$ the number of coefficients in the full model, SSE is the residual sum of squares error and SSR is the regression sum of squares error.

$$\text{SSE} = \sum_{i=1}^n (y_i - \bar{y}_i)^2 \quad (\text{D.3})$$

$$\text{SSR} = \sum_{i=1}^n (\bar{y}_i - \bar{y})^2 \quad (\text{D.4})$$

As seen in figure D.1, the parameter with the highest partial f value is added to the regression model if $f > F_{crit}(1, n - k - 2)$ for a desired significance level (α). F_{crit} is found by solving the F-distribution's cumulative density function for a value of $(1 - \alpha)$ and the pre-ascribed constants.

After a parameter has been added, the algorithm checks that all previously added parameters have maintained their significance. This is done by finding the partial f statistic of removing one parameter at a time. If the minimum value found is lower than the cutoff value given by $F_{crit}(1, n - k - 1)$, that parameter is removed.

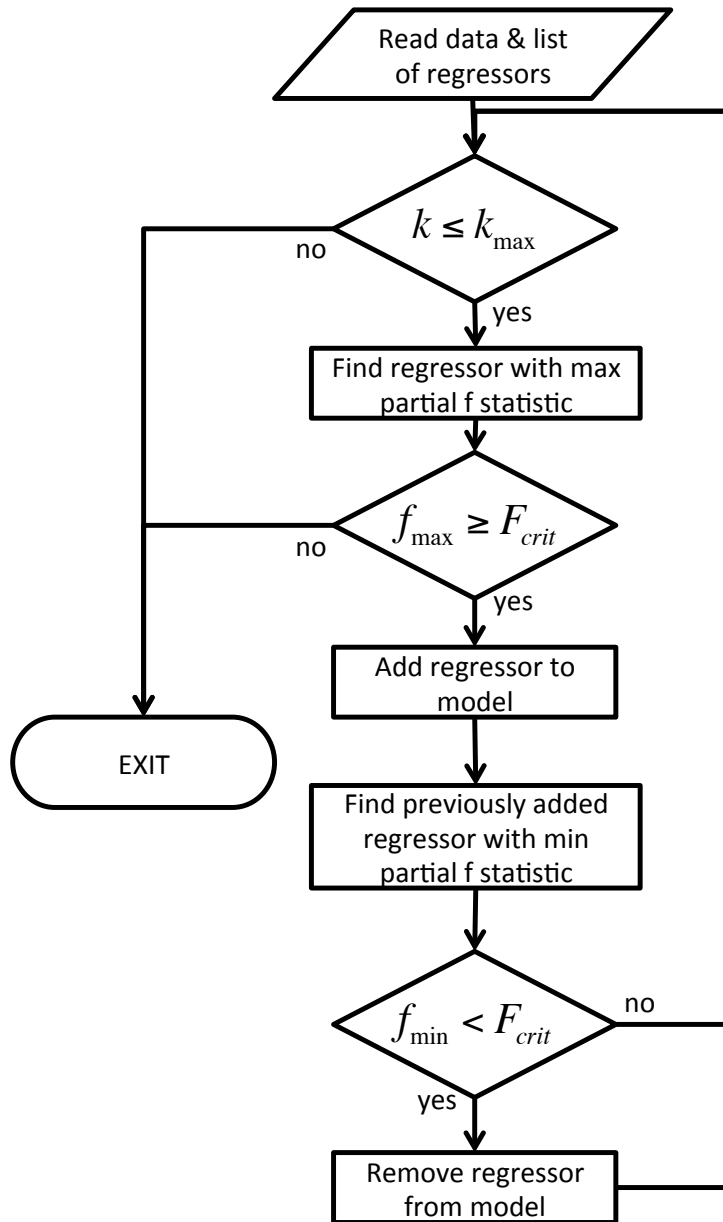


Figure D.1: Flowchart of stepwise regression method used

Appendix E

Hydrodynamic Methods Applicability Range

The hydrodynamic resistance & trim estimation models were generated using experimental data for single and twin step hulls. Some hulls without an afterbody were also used. The strict applicability range of the resulting equations in the following tables.

Parameter	Range		Parameter	Range	
	min	max		min	max
$C_v^2/C_\Delta^{1/3}$	1.501	14.999	C_Δ	0.096	3.077
τ	-1.230	19.900	$\sqrt{2i_e}$	2.1623	12.378
L_f/B	2.279	8.560	$\sec \beta_1$	1.015	1.145
dL'_1	0.857	4.711	$\tan \delta_{k1}$	0.000	0.158
h'_1	0.007	1.0286	$\sec \beta'_2$	1.014	1.252
dL'_2	0.372	2.969	$\tan \delta_{k2}$	0.141	0.460
h'_2	0.015	1.303	$\sec \beta'_3$	1.047	1.252
x_{st}/B	0.000	4.182	z_{st}/B	0.200	2.200

Table E.1: Range of parameters used in building the displacement range models

Parameter	Range	
	min	max
$\sqrt{C_\Delta}/C_v$	0.017	0.29987
C_Δ	0.019	2.000
L_f/B	2.279	8.560
$\sec \beta'_1$	1.015	1.155
dL'_1	0.857	4.711
$\tan \delta_{k_1}$	0.000	0.158
h'_1	0.007	1.029
$\sec \beta'_2$	1.014	1.252
dL'_2	0.372	2.969
$\tan \delta_{k_2}$	0.141	0.460
h'_2	0.015	1.303
$\sec \beta'_3$	1.047	1.2521
x_{st}/B	0.000	4.182
z_{st}/B	0.200	2.200

Table E.2: Range of parameters used in building the planing regime models

Appendix F

Porpoising Data Sources

Source Report	Model ID	Source Report	Model ID
ARC R&M-2899	Sunderland Mk.5	ARC R&M-2641	Princess
ARC R&M-2641	Princess Mod N.	ARC R&M-2834	Princess Mode AF
ARC R&M-2834	Princess Mod. AE	ARC R&M-2834	Princess Mod. AI
ARC R&M-2834	Princess Mod. AK	ARC R&M-2708	All results
ARC R&M-2718	SR.A/1	NACA Re-870	180, 180-1
NACA Re-753	101BA,BB,BC	NACA TN-2503	1056-02
NACA TN-2503	1024-01	NACA TN-2503	1057-03,04
NACA TN-2503	1222-01	NACA TN-2503	40BE
NACA RM-L6J10	212	NACA WR-W-67	All XPB2M-1
NACA WR-L-610	Models A, B, C	NACA TN-2297	40 deg. deadrise
NACA TN-1580	224	NACA TN-1453	224-Extened
NACA TN-1980	224J	NACA TN-1828	224-warped
NACA WR-L-400	134-A - 134-H	NACA WR-L-684	134-G-5
NACA WR-L-684	134-F-8	NACA WR-L-684	134-A-11.5
NACA WR-L-684	134-E-12.2	NACA WR-L-306	All
NACA WR-L-475	117	NACA RM-SL53K06	314, 316
NACA RM-SL55J17	316B	NASA TN-D-529	All
NACA WR-W-105	All	NACA TN-1182	207A
NACA TN-1182	N2/41-AI	NACA TN-1182	Shetland
NACA TN-1182	N2/42-Q	NACA TN-1182	FEH

Source Report	Model ID	Source Report	Model ID
NACA TN-1182	294-79	NACA TN-1182	406
NACA RM-SL9K14	246	NACA RM-L6I26	161, 161A-1
NACA RM-L6I26	165, 165A-1	NACA RM-L5I28	203C-1
NACA RM-L6I20	All 208		

Appendix G

Porpoising Methods Applicability Range

The porpoising models were generated using experimental data for single and twin step hulls. The strict applicability range of the resulting equations is given in the following figure & table.

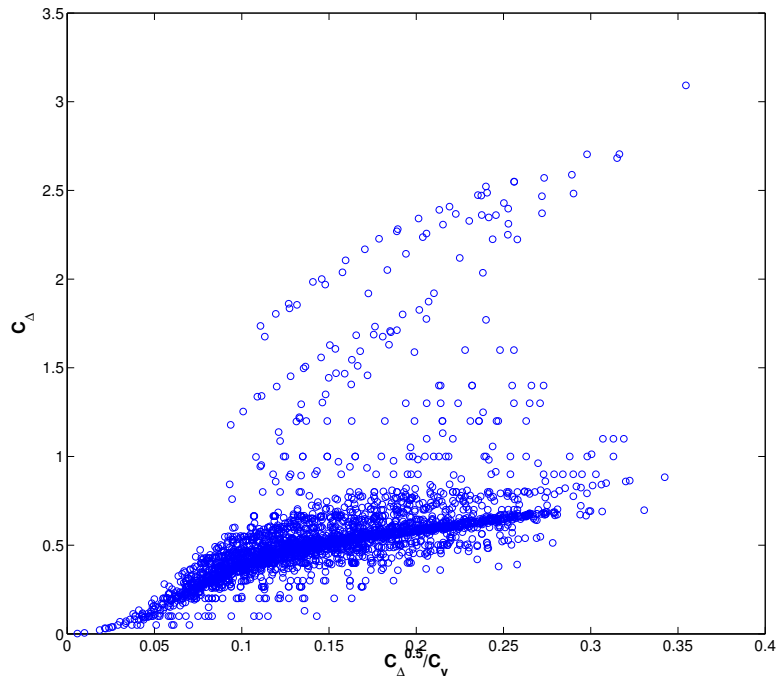


Figure G.1: Applicability range for the porpoising prediction method for hydrodynamic lift coefficient vs beam loading

Parameter	Range	
	min	max
$\sqrt{C}\Delta/C_v$	0.006	0.355
C_Δ	0.002	3.0919
$d\beta/dB$	0.000	25.310
L_f/B	2.820	8.600
$\cos \beta_1$	0.766	0.985
dL'_1	1.540	9.400
$\tan \delta_{k_1}$	0.052	0.213
h_1/B	0.010	0.240
$\sin \delta_{s_1}$	0.4554	1.000
$\cos \beta_2$	0.643	0.985
dL'_2	0.740	3.460
$\tan \delta_{k_2}$	0.112	0.424
h_2/B	0.034	1.839
$\sin \delta_{s_2}$	0.134	0.400
$\cos \beta'_3$	0.70711	0.9962
M'_q	0.000	3.476
Z'_θ	0.000	0.816

Table G.1: Range of parameters used in building the porpoising models

Appendix H

Detailed Weight & Balance Estimation Methods

This appendix contains the equations and methods used to estimate the individual weight and balance contribution of each aircraft component. The wing weight was estimated using the iterative method presented in appendix C of Torenbeek [93]. The wing dimensions used are those of the equivalent straight tapered wing obtained by considering the wing planform outboard of the cabin ($y > y_{le}$) only. The wing's centre of gravity is assumed to be at the 35% semi span location, 70% of the distance between the forward and aft spars. The moment of inertia contribution of the wing, is calculated analytically for the given sweep, taper ratio and dihedral, based on the assumption that the a constant thickness to chord ratio and density are maintained along the span. The weight, balance and inertial contributions of leading and trailing edge devices are also estimated using the aforementioned methodologies.

The weight of fins is estimated using the GD method, as presented by Roskam [72]. The component balance and inertial characteristics are found in the same way as for the wing. For the remainder of this section, the equations referred to can be found in this reference, unless specifically stated otherwise.

The dry engine weight is estimated using the equation given in section 6.3. The nacelle weight is estimated using the GD method and its centre of gravity is assumed to lie at 40% of the nacelle length, forward of the engine fan. If thrust reversers are used, a weight penalty of 18% of the engine dry weight is added. The weight of engine controls is also based on the GD method.

All fuel tanks are assumed to be integral to the structure and their weight is obtained using the Torenbeek method. The fuel system weight is assumed to act at 20% of the straight line distance between the fuel tank and the aircraft's engines. The APU weight is estimated using the method given by Torenbeek [93], in terms of the total cabin bay volumes.

The weight of the flight control system is estimated using the GD methods, based on the aircraft's MTOW. The weight is distributed among the various control surfaces based on their relative areas. The pressurisation and air conditioning system weight is estimated using the GD methods, based on the number of passengers and cabin volume. The system's centre of gravity is assumed to lie close to the location where the PACKs have been situated. The PACK location is determined based on the availability of space, primarily placed underneath the cabin, forward of any under-cabin cargo bays, or otherwise on either side of the cabin, within the wing fairing. The weight of potable water tanks is given by Torenbeek and depends on the aircraft's range. They are placed under the cabin floor, wherever free volume can be found.

The following weights are all found using the methods presented by Torenbeek [93]. The weight and balance contribution of furnishings and the various cabin systems is found by considering the allocation of these components between the individual cabin bays. The weight of lavatories, galleys and the oxygen and electrical systems is based on the weight per unit or per passenger and depends upon the expected aircraft range. The equations given by Torenbeek are also used to determine the weight of floor coverings, wall covers and bins, based on the volume of each individual cabin bay. The weight of seats is distributed between the various cabin bays based on the number of passengers each bay is capable of housing, as calculated during the cabin packaging. The weight of an individual seat and armrest is a predefined user input. Similarly, the weight of material lining the cargo bays is estimated based on each bay's volume.

Staircases are assumed to weigh about 30kg per running meter. The Torenbeek method is used to estimate the weight of avionics, situated underneath the cockpit floor, and the cockpit furnishing. Finally the weight of paint is estimated as 0.6% of the MTOW found.