Development of Interactive Aircraft Design Software for Use in Problem-Based Learning

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Dedication

I would like to dedicate this thesis to:

The Souls of My Mother and Father,

My Lovely Wife,

&

My Smart Sons

Acknowledgements

First of all, I am very grateful to my God, the Most Merciful, the Most Compassionate, for giving me the opportunity, courage, support, and inspiration to undertake this research.

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Abstract

In the last ten years or so, many interactive aircraft design software packages have been released into the market. One drawback of these packages is that they assume prior knowledge in the field of aircraft design. Also, their main purpose being the preliminary aircraft design in a commercial environment, and are not intended for instructional use. Aircraft Design is an iterative process, and the students in the formative years of training must realise that one year of study is not enough to embrace all the necessary underlying concepts in this field. Most universities present the aircraft design as a classical Problem-Based Learning scenario, where students work in groups, with the group size varying between 5 and 8 students., each with a designated role, to carry out a specific task. The students work through the classical process of preliminary design based largely on textbook methods. Therefore, the need for a preliminary design tool (software) that helps the students to understand, analyse, and evaluate their aircraft design process exists.

The developed software does everything that is needed in the preliminary design environment. Students are interactively guided through the design process, in a manner that facilitates lifelong learning. Comprehensive output is provided to highlight the "what if scenarios".

The software consists of many modules such as input (user interface), weight estimation, flight performance, cost estimation, take-off analysis, parametric studies, optimisation, and dynamic stability. Due to the large number of input design variables, a full interactive Graphical-User-Interface (GUI) is developed to enable students to evaluate their designs quickly. Object-Oriented-Programming (OOP) is used to create the GUI environment.

The stability and control derivatives computed in this work are largely based on analytical techniques. However, a facility is provided in the software to create the data input file required to run a software package produced by USAF, called DATCOM, that enables computation of the dynamic stability and control derivatives that can be ultimately used in flight simulation work.

Amongst all the variables used in aircraft design, aircraft weight is the most significant. A new weight estimation module has been developed to increase the accuracy of estimation to better than 5%. Its output results agree very favourably with the published data of current commercial aircraft such as Airbus and Boeing. Also, a new formula is proposed to estimate the engine weight based on its thrust in the absence of the data available with high degree of accuracy.

In order to evaluate the effectiveness of the design under consideration, a comprehensive methodology has been developed that can predict the aircraft price as a function of aircraft weight. The Direct Operating Cost (DOC) is also calculated using methods proposed by ATA, NASA, and AEA.

Finally, a walk-through of two case studies are presented, one for large transport aircraft and other for small business jet, to show how typical undergraduate students will proceed with the design and to demonstrate the effectiveness of the developed software.

Declaration

I confirm that the work submitted is my own and that appropriate credit has been given where reference has been made to the work of others.

The following papers have been published and parts of their materials are included in the thesis:

Published Papers:

[1] Al-Shamma, O., and Ali, R., "Interactive Aircraft Design for Undergraduate Teaching", Applied Aerodynamic Conference: Modelling & Simulation in Aerodynamic Design Process, Royal Aeronautical Society, Bristol, UK, 17-19 July, 2012.

[2] Al-Shamma, O., and Ali, R., "*Aircraft Weight Estimation in Interactive Design Process*", 72nd Annual International Conference on Mass Properties, Society of Allied Weight Engineers, 18-23 May, 2013.

Under Preparation

[3] Al-Shamma, O. & Ali, R., *Cost Estimation Models in Transport Aircraft Design,* 2013, The Aeronautical Journal, Royal Aeronautical Society

[4] Al-Shamma, O. & Ali, R., *Transport Aircraft Weight Estimation as a Function of Range and Number of Passengers***,** 2013, The Aeronautical Journal, Royal Aeronautical Society

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List of Symbols and Abbreviations

a. Symbols (Capitals)

$A_{x_w} = exposed\ wing\ aspect\ ratio$

$A_{ht} = horizontal\ tail\ aspect\ ratio$

$A_{vt} = vertical\ tail\ aspect\ ratio$

$A_w = wing\ aspect\ ratio$

$B_{FL} = balanced\ field\ length$

$C_{AC} = aircraft\ cost$

$C_{AF} = airframe\ cost$

$C_{AFc} = airframe\ cost\ per\ kilogramme$

$C_{(al)_{kh}} = airframe\ labour\ manhours\ per\ flight\ hour$

$C_{(al)_{kc}} = airframe\ labour\ manhours\ per\ flight\ cycle$

$C_{(am)_{kh}} = airframe\ material\ cost\ per\ flight\ hour$

$C_{(am)_{kc}} = airframe\ material\ cost\ per\ flight\ cycle$

$C_{(el)_{kh}} = engine\ labour\ manhours\ per\ flight\ hour$

$C_{(el)_{kc}} = engine\ labour\ manhours\ per\ flight\ cycle$

$C_{(em)_{kh}} = engine\ material\ cost\ per\ flight\ hour$

$C_{(em)_{kc}} = engine\ material\ cost\ per\ flight\ cycle$

$C_{L_{AC}} = aircraft\ lift\ coefficient$

$C_{L_{AC-T}} = aircraft - less - tail\ lift\ coefficient$

$C_{L_{ht}} = horizontal\ tail\ lift\ coefficient$

$C_{L_{nac}} = nacelle\ lift\ coefficient$

$C_{L_w} = lift\ curve\ slope\ coefficient$

$C_{L_d} = ideal\ lift\ coefficient$

$C_{L_{max}} = maximum\ lift\ coefficient$

$C_{L_{mtl}} = aircraft\ maximum\ lift\ coefficient\ at\ landing$

$C_{L_{v2}} = aircraft\ lift\ coefficient\ at\ v2$

$C_{al} = airframe\ labour\ maintenance\ cost\ per\ flight$

$C_{am} = airframe\ maintenance\ cost\ per\ flight$

$C_{amm} = airframe\ material\ maintenance\ cost\ per\ flight$

$C_{bur} = maintenance\ burden\ cost$

$C_{cc} = cabin\ crew\ cost$

$C_{dp} = depreciation\ cost\ per\ flight$

$C_{d0_{ht}} = horizontal\ tail\ zero - lift\ drag$

$C_{d0_{AC}} = aircraft\ zero - lift\ drag$

$C_{d0_{fus}} = fuselage\ zero - lift\ drag$

$C_{d0_{int}} = wing - fuselage\ interference\ zero - lift\ drag$

$C_{d0_{nac}} = nacelle\ zero - lift\ drag$

$C_{d0_{vt}} = vertical\ tail\ zero - lift\ drag$

$C_{d0_{w}} = wing\ zero - lift\ drag$

$C_{d_{AC}} = aircraft\ lift\ drag$

$C_{di_{ht}} = horizontal\ tail\ lift\ induced\ drag$

$C_{di_{AC}} = aircraft\ lift\ induced\ drag$

$C_{di_{fus}} = fuselage\ lift\ induced\ drag$

$C_{di_{int}} = wing - fuselage\ interference\ lift\ induced\ drag$

$C_{di_{nac}} = nacelle\ lift\ induced\ drag$

$C_{di_{vt}} = vertical\ tail\ lift\ induced\ drag$

$C_{di_{w}} = wing\ lift\ induced\ drag$

$C_{el} = engine\ labour\ maintenance\ cost\ per\ flight$

$C_{em} = engine\ maintenance\ cost\ per\ flight$

$C_{emm} = engine\ material\ maintenance\ cost\ per\ flight$

$C_{eng} = engine\ cost$

$C_{engc} = engine\ cost\ per\ one\ pound\ thrust$

$C_{f_{ht}} = horizontal\ tail\ skin\ friction\ coefficient$

$C_{f_{fus}} = fuselage\ skin\ friction\ coefficient$

$C_{f_{nac}} = nacelle\ skin\ friction\ coefficient$

$C_{f_{vt}} = vertical\ tail\ skin\ friction\ coefficient$

$C_{f_w} = wing\ skin\ friction\ coefficient$

$C_{fc} = flight\ crew\ cost$

$C_{fu} = fuel\ cost\ per\ gallon$

$C_{fuel} = fuel\ cost\ per\ flight$

$C_{grd} = ground - handling\ cost\ per\ flight$

$C_{ins} = insurance\ cost\ per\ flight$

$C_{int} = interest\ cost\ per\ flight$

$C_{lf} = landing\ fees\ per\ flight$

$C_{lr} = labour\ cost\ per\ hour$

$C_{m_{AC}} = aircraft\ pitching\ moment\ coefficient$

$C_{m_{fus}} = fuselage\ pitching\ moment\ coefficient$

$C_{m_w} = wing\ pitching\ moment\ coefficient$

$C_{maint} = total\ maintenance\ cost$

$C_{nav} = navigational\ charges\ per\ flight$

$C_{oil} = oil\ cost\ per\ gallon$

$D_{fus} = fuselage\ diameter$

$D_{nac} = nacelle\ diameter$

$E_{bpr} = engine\ bypass\ ratio$

$E_{oapr} = overall\ compressor\ ratio$

$F_i = international\ salary\ premium\ (= 1\ for\ domestic, = 1.1\ for\ international\ flights$

$G_{seg2} = aircraft\ second\ segment\ climb\ gradient$

$H_t/H_v = 0\ for\ conventional\ tail, 1\ for\ T - tail$

$L_{FL} = landing\ field\ length$

$M_{ht} = horizontal\ tail\ moment$

$M_{aw} = aft\ window\ seats\ payload\ moment$

$M_e = total\ empty\ aircraft\ moment$

$M_{eng} = engine\ moment$

$M_{fuel} = fuel\ moment$

$M_{furn} = furnishings\ moment$

$M_{fus} = fuselage\ moment$

$M_{fw} = forward\ window\ seats\ payload\ moment$

$M_{nac} = nacelle\ moment$

$M_{pay} = full\ payload\ moment$

$M_{sur} = surface\ controls\ moment$

$M_{sys} = systems\ moment$

$M_{uc} = undercarriage\ moment$

$M_{vt} = vertical\ tail\ moment$

$M_w = wing\ moment$

$N_c = number\ of\ compressor\ stages\ per\ engine$

$N_{eng} = number\ of\ engines$

$N_{fl_{att}} = number\ of\ flight\ attendants$

$N_{fl_{crew}} = number\ of\ flight\ crew$

$N_{gust} = gust\ load\ factor$

$N_{ma} = Mach\ number$

$N_{manu} = manoeuvre\ load\ factor$

$N_{pas} = number\ of\ passengers$

$N_{rey} = Reynolds\ number$

$N_{row} = number\ of\ rows$

$N_{seat/row} = number\ of\ seats\ per\ row$

$N_{sf_{eng}} = engine\ scale\ factor$

$N_{ult} = ultimate\ load\ factor$

$P_{af} = airframe\ life\ (Liebeck\ assumes\ 15\ years)$

$P_{eng} = engine\ life\ (Liebeck\ assumes\ 15\ years)$

$R_{b_{f_i}/b} = flap\ inboard\ span\ to\ wing\ span\ ratio$

$R_{b_{f_o}/b} = flap\ outboard\ span\ to\ wing\ span\ ratio$

$R_c = rate\ of\ climb$

$R_{c_f/c_w} = flap\ chord\ to\ wing\ chord\ ratio$

$R_{l_{3_{fus}}/D_{fus}} = tail\ cone\ to\ fuselage\ diameter\ ratio$

$R_{p_w/l_{fus}} = wing\ position\ to\ fuselage\ length\ ratio$

$R_{ins} = insurance\ rate$

$R_{int} = interest\ rate$

$S_{af} = airframe\ spares\ (assume\ 0.06 \times airframe\ cost)$

$S_e/S_{ht} = elevator\ area\ to\ horizontal\ area\ \approx 0.2$

$S_{eng} = engine\ spares\ (assume\ 0.23 \times engine\ cost)$

$S_f = flap\ area$

$S_{ht} = horizontal\ tail\ area$

$S_{vt} = vertical\ tail\ area$

$S_w = wing\ reference\ area$

$S_{w_{flap}} = flapped\ wing\ area$

$S_{w_{fus}} = fuselage\ wetted\ area$

$S_{w_{nac}} = nacelle\ wetted\ area$

$S_{x_w} = exposed\ wing\ area$

$T_{AC} = aircraft\ thrust$

$T_{eng} = engine\ thrust$

$U = aircraft\ utilisation\ (hours/year)$

$V_{AC} = aircraft\ velosity$

$V_{ht} = horizontal\ tail\ volume\ coefficient$

$V_{vt} = vertical\ tail\ volume\ coefficient$

$V_{dive} = designed\ dive\ speed$

$V_{sl} = landing\ stall\ speed$

$V_{st} = takeoff\ stall\ speed$

$X_{cg_e} = empty\ aircraft\ CG\ position$

$X_{cg_{fe}} = fueled\ empty\ aircraft\ CG\ position$

$X_{cg_{ffa}} = fueled\ full\ payload\ aft\ CG\ position$

$X_{cg_{fff}} = fueled\ full\ payload\ forward\ CG\ position$

$X_{cg_{fsa}} = fueled\ window\ seats\ payload\ aft\ CG\ position$

$X_{cg_{fsf}} = fueled\ window\ seats\ payload\ forward\ CG\ position$

$X_{cg_{zfa}} = zero - fuel\ full\ payload\ aft\ CG\ position$

$X_{cg_{zff}} = zero - fuel\ full\ payload\ forward\ CG\ position$

$X_{cg_{zsa}} = zero - fuel\ window\ seats\ payload\ aft\ CG\ position$

$X_{cg_{zsf}} = zero - fuel\ window\ seats\ payload\ forward\ CG\ position$

b. Symbols (Non-Capitals)

$b_{f_i} = flap\ inboard\ span$

$b_{f_o} = flap\ outboard\ span$

$b_{x_w} = exposed\ wing\ span$

$b_{ht} = horizontal\ tail\ span$

$b_{vt} = vertical\ tail\ span$

$b_w = wing\ span$

$c_{r_{ht}} = horizontal\ tail\ root\ chord$

$c_{r_{vt}} = vertical\ tail\ root\ chord$

$c_{r_w} = wing\ root\ chord$

$c_{t_w} = wing\ tip\ chord$

$\bar{c} = mean\ aerodynamic\ chord$

$\bar{c}_{ht} = horizontal\ tail\ mean\ geometric\ chord$

$\bar{c}_{vt} = vertical\ tail\ mean\ geometric\ chord$

$\bar{c}_w = wing\ mean\ geometric\ chord$

$l_{1_{fus}} = nose\ cone\ length$

$l_{2_{fus}} = cabin\ length$

$l_{3_{fus}} = tail\ cone\ length$

$l_{fac} = distance\ from\ fuselage\ nose\ to\ wing\ centre\ of\ pressure$

$l_{fqc} = distance\ from\ fuselage\ nose\ to\ wing\ quarter\ root$

$l_{nac_{le_{\bar{c}/4}}} = distance\ from\ nacelle\ leading\ edge\ to\ forward\ of\ wing\ quarter - chord\ line$

$l_{p_w} = distance\ from\ nose\ to\ the\ leading\ edge\ of\ wing\ at\ the\ centre\ line$

$l_{ht} = horizontal\ tail\ arm$

$l_{eng} = engine\ length$

$l_{fus} = fuselage\ length$

$l_{nac} = nacelle\ length$

$l_{vt} = vertical\ tail\ arm$

$m_{AC} = aircraft\ mass$

$m_T = tail\ mass$

$m_{acic} = air\ conditioning\ and\ anti - icing\ mass$

$m_{apu} = auxiliary\ power\ unit\ mass$

$m_{apu_{dry}} = dry\ unstalled\ auxiliary\ power\ unit\ mass$

$m_e = aircraft\ empty\ mass$

$m_{e-e} = aircraft\ empty\ mass\ minus\ engines\ mass\ (lbs)$

$m_{ele} = electrical\ system\ mass$

$m_{eng} = engine\ mass$

$m_{fl_{att}} = flight\ attendants\ mass$

$m_{fl_{crew}} = flight\ crew\ mass$

$m_{fuel} = design\ fuel\ mass$

$m_{furn} = furnishings\ mass$

$m_{fus} = fuselage\ mass$

$m_{hyd} = hydraulics\ and\ pneumatics\ mass$

$m_{ins} = instruments\ and\ avionics\ mass$

$m_{lan} = landing\ aircraft\ mass$

$m_{mg} = main\ gear\ mass$

$m_{nac} = total\ nacelles\ mass$

$m_{ng} = nose\ gear\ mass$

$m_{oe} = aircraft\ operating\ empty\ mass$

$m_{op\_it} = operating\ items\ mass$

$m_{oxy} = oxygen\ system\ mass$

$m_{pay} = payload\ mass$

$m_{pnt} = paint\ mass$

$m_{pro} = total\ propulsion\ group\ mass$

$m_{pro\_sys} = propulsion\ system\ mass$

$m_{sur} = surface\ controls\ mass$

$m_{sys} = systems\ mass$

$m_{to} = maximum\ takeoff\ weight$

$m_{uc} = undercarriage\ mass$

$m_w = wing\ mass$

$m_{wpay} = window\ seats\ payload\ mass$

$m_{zf} = zero\ fuel\ mass$

$r_{ht} = horizontal\ tail\ roughness\ factor$

$r_{fus} = fuselage\ roughness\ factor$

$r_{int} = wing - fuselage\ interference\ roughness\ factor$

$r_{nac} = nacelle\ roughness\ factor$

$r_{vt} = vertical\ tail\ roughness\ factor$

$r_w = wing\ roughness\ factor$

$s_l = main\ mission\ stage\ length$

$t_B = block\ time\ in\ hours$

$t_{c_{ht}} = horizontal\ tail\ thickness\ ratio$

$t_{c_{vt}} = vertical\ tail\ thickness\ ratio$

$t_{c_w} = wing\ thickness\ ratio$

$t_f = flight\ time\ in\ hours$

$v = aircraft\ velocity$

$w_{inc} = wing\ incidence\ angle$

$w_{TCRT} = wing\ root\ thickness$

$w_{TCRTR} = wing\ root\ thickness\ to\ tip\ thickness\ ratio$

$x_{l_{eng}} = engine\ span - wise\ distance$

c. Greek Symbols

$\Lambda = wing\ sweep\ angle\ at\ quarter\ chord$

$\Lambda_h = horizontal\ tail\ sweep\ angle$

$\Lambda_v = vertical\ tail\ sweep\ angle$

$\delta_{flan} = maximum\ flap\ deflection\ at\ landing$

$\delta_{fto} = maximum\ flap\ deflection\ at\ takeoff$

$\lambda_{ht} = horizontal\ tail\ taper\ ratio$

$\lambda_{vt} = vertical\ tail\ taper\ ratio$

$\lambda_w = wing\ taper\ ratio$

$\rho_f = fuel\ density\ (lbs/gal)$

$\alpha = angle\ of\ attack\ of\ aircraft$

$\pi = pi$

d. Abbreviations

AEA = Association of European Airliners

AEO = All Engines Operating

AOA = Angle Of Attack

APU = Auxiliary Power Unit

ATA = Air Transportation Association of America

BFL = Balanced Field Length

CG = Centre of Gravity

CPI = Consumer Price Index

DATCOM = Data Compendium

DOC = Direct Operating Cost

EW = Empty Weight

GUI = Graphical User Interface

IOC = Indirect Operating Cost

ISA = International Standard Atmosphere

LFL = Landing Field Length

MTOW = Maximum Take-Off Weight

OEI = One Engine Inoperative

OEW = Operating Empty Weight

OOUI = Object-Oriented User Interface

RAE = Royal Aircraft Establishment

SMC = Seat Mile Cost

TOC = Total Operating Cost

WAT = Weight Altitude Temperature

WIMP = Windows, Icons, Menus, and Pointers

iADS = Interactive Aircraft Design Software

Chapter ONE

Introduction

1.1 History background

Aircraft Design is both an Art and Science [1]. One can argue the case for this statement. The word *Art* does not mean or imply connection with the artistic elements of aircraft design, here the implicit meaning is understood to be "*A work of art is an artefact of a kind created to be presented to a set of persons, the members of which are prepared in some degree to understand an object which is presented to them"* [2]. Since an aircraft can be regarded as an artefact, the interpretation that it is an art is correct. On the other hand, *Science* is universally defined as, "*1- the systematic observation of natural events and conditions in order to discover facts about them and to formulate laws and principles based on these facts. 2- the organised body of knowledge that is derived from such observations and that can be verified or tested by further investigation*" [3]. So yes, *Aircraft Design* is both an *Art* and a *Science*.

The proliferation of the human race mean they employed the land and water as a medium through which it was able to reach to remote areas of this world, and for centuries dreamt of somehow acquiring the ability to be able to use the third medium, "*Air*". This meant emulating the birds that grace our skies. Knowledge is incremental, and often builds upon the findings of our predecessors. According to a commonly known myth, an ancient Greek legend Daedalus, an engineer who was imprisoned by King Minos, with his son Icarus, made wings of wax and feathers. Daedalus flew successfully from Crete to Naples, but Icarus, tried to fly too high and flew too near to the sun. The wings held together with wax melted and Icarus fell to his death in the ocean. If one assumes that the myth has an element of truth, then one can only say that the flight was doomed from the start, as the engineers had little or no scientific knowledge as regards the material properties or of the theories that eventually evolved that outlined the basic principles of flight and wax melting with increasing temperature.

Many advances over centuries from the kite in 400BC, to Hero's Aeolipile, the first steam powered rotation device, to Leonardo da Vinci's flapping wing ornithopter to Joseph and Jacques Montgolfier's hot air balloon helped in adding to the knowledge pool that led Sir George Cayley to produce a glider, and in the process, he was able to specify the importance of separating lift and propulsion and the importance of dihedral. He also developed whirling-arm apparatus to measure forces on aerofoils and wings and quantified the importance of camber. Otto Lilienthal also known as the “Father" of hang-gliding, in his experiments discovered the importance of control and developed extensive tables of lift and drag forces based on (flawed) whirling-arm experiments.

One of the most important contributions came from Octave Chanute who took European aeronautical knowledge to the USA and published *Progress in Flying Machines* in 1894. This information ultimately helped the Wright brothers in pursuit of the dream to fly. Wilbur and Orville discovered the importance of 3-axis control (but not stability) and learned to control flight in extensive glider experiments. They discovered errors in Lilienthal’s whirling-arm data and built a wind-tunnel for aerodynamic testing, enabling them to understand lift and drag. They also developed the first theory for propellers and built one that had better than 80% efficiency. It was not until 1903, when the Wright Flyer made history. According to the Fédération Aéronautique Internationale (FAI), Orville Wright made the first sustained, controlled, and powered heavier-than-air manned flight of 120 feet in 12 seconds in North Carolina on December 17, 1903. He continued to improve his plane with his brother Wilbur, which included many changes to control surfaces as well as the wing. On October 5, 1905, the Wright Brothers made *Flyer III* which was the first practical airplane that flew in the air for a distance of 24 miles in 39 minutes 23 seconds. Santos-Dumont, from Brazil, made other contributions to the field of aircraft design. He added movable surfaces, the precursor to *ailerons*, between the wings in an effort to gain more lateral stability. Santos-Dumont's final design was the *Demoiselle* monoplane (No.s 19 to 22) in 1909. In 1911, the first transcontinental flight across the U.S. was completed by Calbraith Rodgers. His flight from New York to California took 3 days, 10 hours, and 14 minutes, and was flown in a Wright aircraft [4].

Soon after, aviation science gathered pace and the English Channel was crossed in 1909, with the DC3 entering active service in 1933. The DC3 had retractable landing gear, fully cantilevered wing, monocoque construction, wing flaps and low-drag engine cowling. These features are also found in the present day aircraft.

The start of World War II saw great strides being made in order to gain air superiority, and very efficient aircraft designs were evolved mainly to satisfy diverse mission objectives. The early piston driven multi-engine to the modern super-jets powered by efficient turbo-fan engines have quite a few things in common, and needless to say they all share similar aerodynamic configuration. One can safely say that the basic configuration that we see today has resulted from extensive scientific probing and flight testing.

After World War II and by 1947 all the basic technology needed for aviation had been developed: jet propulsion, aerodynamics, radar, etc. Civilian aircraft orders rapidly grew from 6,844 in 1941 to 40,000 by the end of 1945. With all the new technologies developed by that time, airliners were larger, faster, and featured pressurised cabins. New aerodynamic designs, materials, and power plants resulted in high-speed turbo-fan/jet airplanes. These planes are able to fly supersonically and make transoceanic flights regularly [5].

The advances in technology meant that aircraft could fly faster and over greater distances. The design of aircraft is a function of the requirements. For instance for commercial aircraft a top cruise speed of around 600 knots is almost a maximum which is at the limit of the sub-sonic flight regime. Transonic and supersonic flight regimes require a radical change in the wing design that requires extremely low drag. Whilst for military applications these designs are plausible, for commercial applications, these become cost ineffective, as the commercial operations are cost sensitive and require a very low seat mile cost, a factor that decides profitability of operation. These days, not only is the aircraft required to be commercially viable, but needs to be ecologically friendly.

In order to comply with requirements and regulations, improvements in all technological areas are continually sought. These changes in technology lead to new methods and processes and affect the design of aircraft.

Aircraft Design is truly multidisciplinary, and involves knowledge in propulsion, aerodynamics, materials, structures, flight mechanics, control and numerous other disciplines. In order to serve the needs of the aviation industry, that needs the designers and maintenance engineers, the knowledge base has to be imparted to those who wish to embark upon a career in aviation industry. The question that arises is, “how best to teach this art and science of Aircraft Design”? Before we can commit to defining the associated curriculum that will enable the knowledge transfer, the process of aircraft design needs to be understood first.

1.2 Aircraft design process

Design practice consists of three distinct phases. From the design viewpoint, these are: Conceptual, Preliminary, and Detail design phase. It should be noted that these phases are not completely separated and tend to overlap considerably depending on the manufacturer, as shown in **Figure 1-1** [6]:

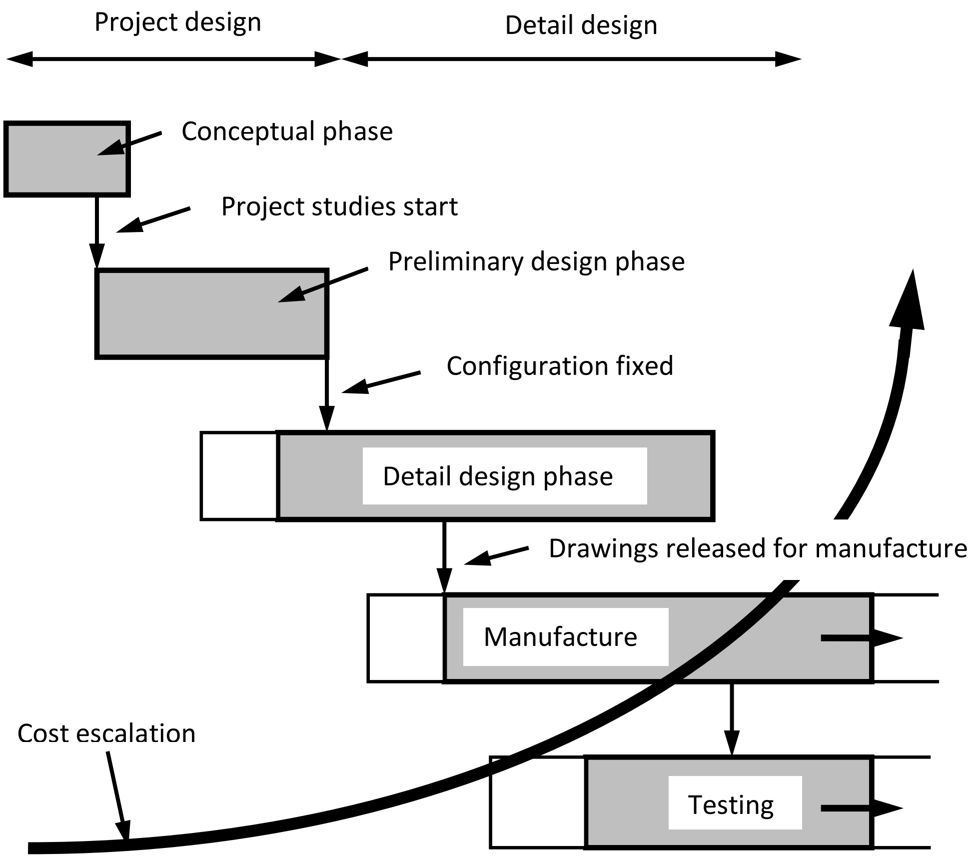

Figure 1-1: Phases of aircraft design process, [6]

The cost increases with each process, and any mistakes in the later stages prove to be very costly. So, it is imperative that most issues are ironed out in the early stages of design.

1.2.1 Conceptual design phase

The first phase, as the name implies, deals with the design at the conceptual level. It starts with a specific set of design requirements usually defined by the customer. Range, seating capacity, and field distances may be an example of these requirements. Other requirements are imposed by airworthiness standards such as FAR [7] and JAR [8]. At this stage, the fundamental tool is “selection”. Therefore, there are not many calculations, but there are variety of evaluations and analysis. This phase is dependent on the past design experience to determine the configuration layouts which are technically feasible and commercially viable. Layout details include aircraft overall geometry, wing and tail configurations, the engine (type, number, and position), undercarriage configuration, control surfaces arrangements, maximum take-off weight estimation, etc. In addition, it may be necessary to investigate the feasibility of the novel concepts and technologies that may be used in the design under consideration.

1.2.2 Preliminary design phase

In this phase, as the name implies, all the determined parameters are subject to change, as the design process is iterative. However, accuracy of the determined results can affect the aircraft design, and it would suffice to say, the designers look towards achieving an accuracy of better than 5% in their calculations/estimations. All designed layouts are compared and analysed carefully and the one closest to fulfilling the requirements is chosen. A model (mock-up) is usually constructed and tested either physically or visualised using a CAD system. It is preferred to establish a baseline configuration and to perform a series of parametric studies around this layout. These changes continue iteratively until the proposed layout completely satisfies the specifications. For instance, the aim of this phase is to find the best optimum geometry for the aircraft with respect to the commercial requirements. The preliminary design phase ends after the configuration is “frozen” and a decision is taken to proceed to the detail design phase without return.

1.2.3 Detail design phase

Detail design phase is the most expensive and extensive phase in the whole design process. Due to full-scale development, the phase starts to verify and refine the design to a greater level of detail and to produce data necessary for the manufacture of the aircraft. A huge number of drawings and/or computerised CAD files is needed to define the aircraft adequately and to ensure that each item designed represents the best solution in terms of performance, manufacturing costs, and operations. The second part of this phase is called production design. Specialists determine how the aircraft will be fabricated, starting with the smallest and simplest subassemblies and building up to the final assembly process. Some parts are built for test purposes at the beginning of the production phase, and the first aircraft is used for initial flight tests. Ground testing includes the use of wind tunnels, structural specimens, and systems rigs. Flight testing is used to verify the performance and flight characteristics of the actual aircraft. At the completion of all tests, the aircraft will be granted a Certificate of Airworthiness by the national aviation authority. Detail design phase ends with fabricating and testing the first aircraft. **Figure 1-2** summarises the activities of aircraft design phases [9].

There are many excellent textbooks covering aircraft design such as References [10][11][12][13][14]. These books are quite detailed in their presentation of design process and cover all aspects of aeronautical engineering subjects that needed in the aircraft design process. To a novice, the information presented can be daunting, and often confusing, as there are so many permutations of the same variables, and the eventual effect of some choice may not be fully understood.

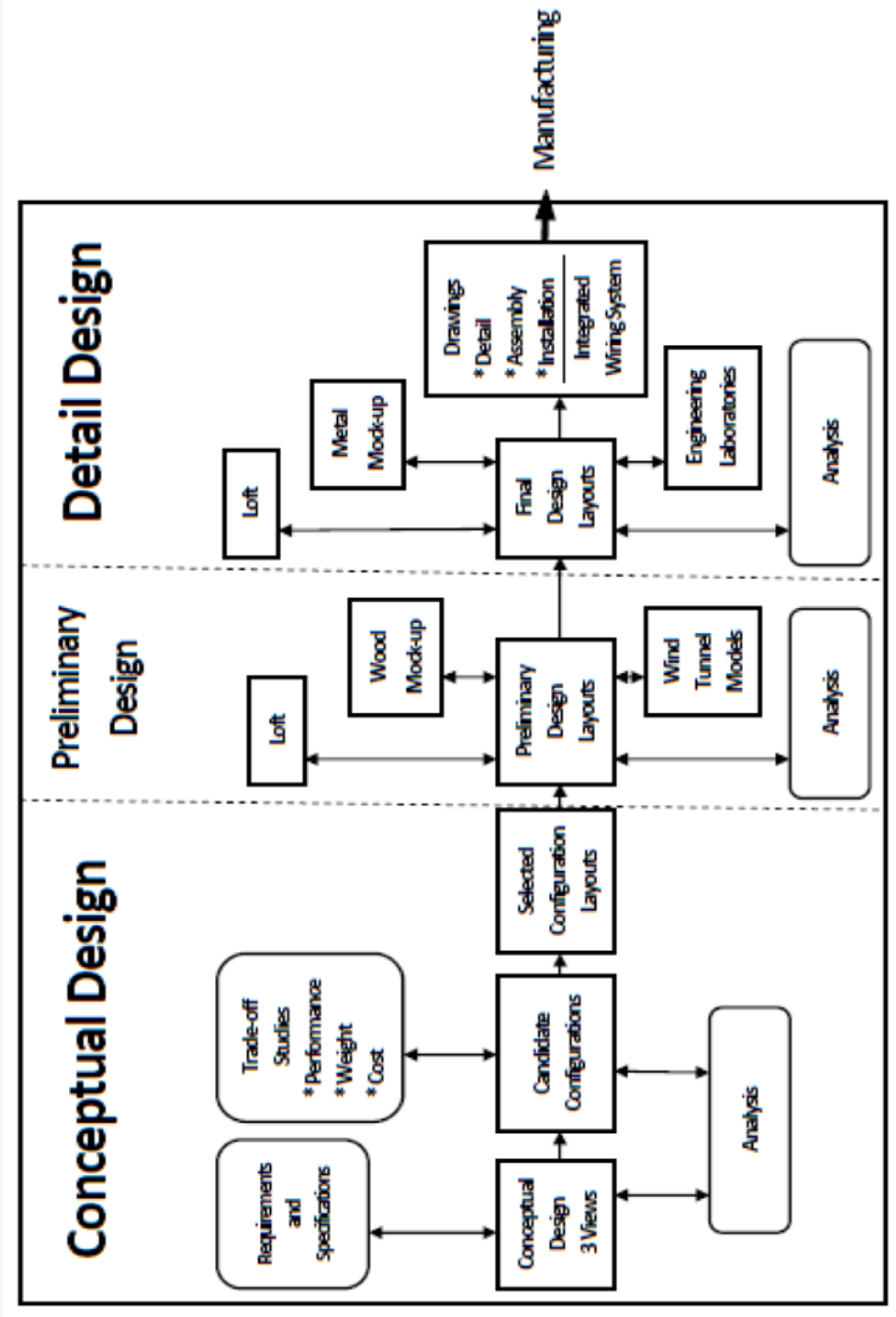

Figure 1-2: Traditional aircraft design phases, [9]

1.3 Teaching aircraft design

1.3.1 Introduction

Aircraft design is a multidisciplinary subject. It requires knowledge in aerodynamics, structures, propulsion, stability and control. Many universities in the United Kingdom teach aspects of aircraft design, as a part of the Aeronautical Engineering curriculum, in the final year of their degree programme. By this stage, the students have acquired basic skills in the fundamental aeronautical science. Aeronautics is defined as the science of operating aircraft. It is concerned basically with predicting and controlling the forces and moments of an aircraft travelling in the air [15]. Engineering is the practice of applying scientific knowledge to the design, construction, and operation of machines and instruments. Engineering design is the process of devising a feasible and efficient solution to some specified needs. It is a complex process of creative and analytical steps, where basic and engineering sciences are applied to convert resources into real products [16]. The designer is required to solve an ill-defined problem to reach the best (or one of the best) among the many possible solutions. The philosophy of aeronautical engineering course is that knowledge, understanding, and skills needed for aircraft operation and design are best acquired through interdisciplinary teaching and demanding application, rather than through a disjointed series of individually assessed modules [17]. Therefore, "Aircraft Design" must be a mandatory part of any aeronautical engineering curriculum. "*It is the corner stone of the aeronautical engineering syllabi. It is recognised as a science that educates the student in synthesis perspective and contributes to create in the future professional an open mind mentality."* [18]. So, the question is: what is the best way to teach students about aircraft design? Many universities follow a Problem-Based-Learning approach.

1.3.2 Problem-based learning

Problem-Based Learning is defined by Barrows [19] as: "*The learning that results from the process of working towards the understanding of resolution of a problem. The problem is encountered **first** in the learning process*". **Problem** is something that cannot be resolved with the current level of knowledge and/or way of thinking about the issues. It is presented as an ill-structured as opposed to well-structured problem. It is characterised as a real-life and authentic, not a hypothetical exercise, messy not tidy, incomplete in the sense of lacking information needed for its solution and iterative in the way that it produces further ideas and learning issues [20].

Problem-Based Learning originated from a curriculum reform by medical faculty at the Case Western Reserve University in the late 1950s. Innovative medical and health science programmes continued to evolve the practice of Problem-Based Learning, particularly the specific small group learning and tutorial process that was developed by medical faculty at McMaster University in Canada. These innovative and forward-looking medical school programs considered the intensive pattern of basic science lectures followed by an equally exhausting clinical teaching program to be an ineffective and dehumanising way to prepare future physicians. Given the explosion of medical information and new technology, as well as the rapidly changing demands of future medical practice, a new mode and strategy of learning was developed that would better prepare students for professional practice. The Problem-Based Learning has spread to over 50 medical schools, and has diffused into many other professional fields including law, economics, architecture, mechanical and civil engineering [21].

The research has found that traditional educational approaches (e.g., lectures) do not lead to a high rate of knowledge retention. Despite intense efforts on the part of both students and lecturers, most material learned through lectures is soon forgotten, and natural problem solving abilities may actually be impaired. The motivation in such traditional classroom environments is also usually low.

One of the greatest advantages of Problem-Based Learning is that students genuinely enjoy the process of learning. The Problem-Based Learning is a challenging programme because students are motivated to learn by a need to understand and solve real problems. The relevance of information learned is readily apparent; students become aware of a need for knowledge as they work to resolve the problems. "*It is vital* *that the problems are engaging, that they "smell real", are interesting and challenging to students. This engagement stimulates further learning and requires research, elaboration, further analysis and synthesis together with decisions and action plans*" [22].

The characteristics of Problem-Based Learning [23] are:

1. *The problem challenges students' existing knowledge, attitudes and competencies, leading them to identify problem new knowledge (or learning issues) needed, and shortcomings that need to be corrected.*
2. *The responsibility and direction of learning is assumed by the students. Faculty members are only there to facilitate students' thinking, learning and group functioning to help them resolve any problem.*
3. *Information mining from various source's, and utilisation of evaluation to analyse what is really useful.*
4. *The process of identifying learning issues and problem-solving is as important as acquiring new knowledge to arrive at the solution.*
5. *Students learn in cooperative teams, where they need to interact and communicate to share knowledge, discuss their understanding and debate conflicting opinion.*
6. *Synthesis of various knowledge and information to arrive at the solution*
7. *Reflection of the students' learning experience.*

Valuable details of Problem-Based Learning have been outlined by Ribeiro [24], Hmelo-Sliver [25], and Newman [26].

1.3.3 Implementation of Problem-Based Learning

Designing an aircraft is a complex and iterative task. To achieve this objective, one or two semesters are not enough time to fully cover all concepts of aircraft design. Most universities present preliminary design projects as in-course assessment (ICA) for this reason. Students are sub-grouped into 5-8 student teams with the lecturer playing the role of facilitator. Teams are given a set of specifications for an aircraft to be designed (i.e. a project). The specifications may include payload, speed, range, take-off, landing performance, or specific mission objective, etc (i.e. requirements of the design). Students start searching with textbook examples, or existing designs, hoping to find a similar solution to their problem that requires only minimal amendments. Actually, these textbook examples are intended to show the way the design process may be applied to those who are starting to undertake aircraft conceptual design for the first time. *"It should be noted that these projects are not meant to provide a "fill in the blank" template to be used by current and future students working on similar design problems, but to provide insight into the process itself"* [27]. In some cases, students may browse the internet searching for a computer code (program) that may help them with the initial variable selection that may meet their design objectives and how best to get final design results. From the faculty view point, students should start the design process through reviewing their previous courses in aerodynamics, structures, propulsion, performance, etc.

Implementation of Problem-Based Learning methodology in aircraft design starts with "Problem Overview". Students have to read and understand the project scenario, organise their ideas, reflect and explain possible issues individually based on the available knowledge. Then, they have to identify the aspects of the problem and needs for research and literature review (learning issues). They are encouraged to do background reading on these issues. Subsequently, they sort the issues and plan when, who, where, how these issues will be investigated. During initial few meetings, they share and explore the gathered knowledge about the learning issues and use it to propose an informed solution to the problem. They may have to restart the cycle if they cannot get a satisfactory solution. The team leader and the role of each individual in the team are decided at the end of this stage.

The second stage is "Conceptual Sketches and Analysis". Students start by producing conceptual sketches of the proposed aircraft. The creative part of the design process, which has no rules and constraints, may produce neither logical nor illogical design alternatives. It is based on uncritical brain storming, observation of nature, and ideas from other engineering disciplines. Thereafter, these sketches are analysed separately depending largely on textbook methods and a single design is adopted by the team. Students may also identify appropriate existing knowledge and more learning issues than those decided in the previous stage. At this stage, facilitators guide the students so that they are on the right track checking and questioning the learning issues identified.

The third stage is “Synthesis and Application”. Each student in the team starts to synthesise his/her own part (component) of the proposed aircraft. The Final solution is an assembly of individual student contributions to the whole aircraft. Information is shared and critically reviewed so that the relevant ones can be synthesised and applied to solve the problem. Facilitators at this stage ensure that the coverage of the problem is sufficient, and continually probe students on accuracy and validity of the information obtained. This can be an iterative process, where students may need to re-evaluate the analysis of the problem, required further learning, reporting and peer learning. The outcome of this stage is the preliminary layout of the proposed aircraft and an integrated report is produced which addresses not only the technical aspects of the design but also the financial viability of the concept [1].

The final stage is “Presentation, Reflection, and Assessment”. The designed aircraft is presented to the class and audience that may consist of practitioners of aircraft design working in industry, followed by probing questions to assess their deeper learning and understanding. Students are also asked to reflect on the content as well as the design. The facilitator helps with integrating knowledge learnt from solving the problem with what they have already, and encourages students to give their opinion on the value and usefulness, of the proposed design, for future learning and application to the work place. The facilitator also summarises crucial principles and concepts, as well as eliminates any doubts that are aired by the students. **Figure 1-3** summarises the stages of implementing Problem-Based Learning process in Aircraft Design.

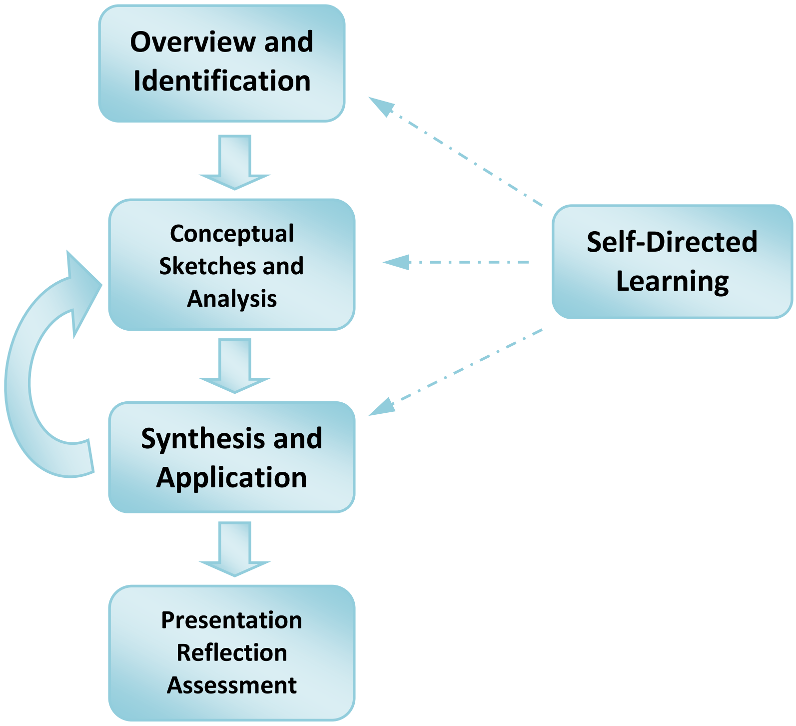

Figure 1-3: Framework of Problem-Based Learning process in Aircraft Design

1.4 Conclusions

The long journey of creation and innovation in aviation history shows that aircraft design is not simple. It is truly a multidisciplinary subject. It hinges on aerodynamics, structures, propulsion, and stability and control. Aircraft design process consists of three phases which are conceptual, preliminary, and detail design. Many universities in the UK, follow Problem-Based Learning approach in teaching aircraft design. Since, the aircraft design process is an iterative process and one year of study is not enough time to fully cover the concepts of the aircraft design. However, at the end of the project, students gain valuable insight into the design process, and this prepares them well to embark upon a career in the aircraft industry. In most universities, students undertake a preliminary design project as an in-course assessment, and pertinent feedback is provided by the facilitators on a weekly basis, to steer the students towards a feasible design.

Chapter TWO

Aircraft Design Software

2.1 Introduction

Computers have become a mainstream part of most academic engineering curriculum. The integration of computer software into Problem-Based Learning can encourage students to learn effectively, deeper, and with much more interest than before, by varying the ways in which information is delivered [28]. If so, is there any aircraft design software that can be used for undergraduate teaching? First of all, there is no such software that takes the design objectives (requirements) and produces a viable aircraft design. Secondly, many young students, who have grown to expect right and wrong answers to problems, are uncomfortable to know that aircraft design has no unique answer. During the conceptual design phase the student has to determine the design manually. The need for a preliminary design software tool that enables the student to understand and analyse the design becomes important.

Now, the question is more specifically: is there any preliminary design software available in the market for teaching? What are its characteristics? To answer the question, existing software is reviewed first, in order to evaluate its effectiveness for the purposes of undergraduate teaching. If not, then a case for developing such software needs to be answered.

2.2 Early aircraft design software

In the mid-sixties and the seventies, several programs were developed. SYNAC II, an optimised program coded in Fortran IV, was developed by Lee, et al [29] at 1967. "*The fundamental philosophy of the program can be summarised as follows: It is intended to replace completely the usual manual process of aircraft synthesis which is associated with basic configuration selection*" [29]. It consisted of input, geometry, weight, propulsion, performance and optimiser modules. The major function of SYNAC was the computation of flight performance. The key drivers were aerodynamic, propulsion forces, and the weight, as a result of the requirement set of the proposed aircraft. Due to the limited computational power at that time, the basic synthesis execution time was one minute per problem without optimisation and no graphic output obtained.

At the same time, Lockheed Georgia [30] produced a computer-aided aircraft design (CAAD) program which included the use of computer graphics and interactive user interface. The program was used until the nineties. "*It is being developed as a prototype covering the aircraft preliminary design and performance estimation process*", [30]. The multivariate optimisation program developed by Kirkpatrick & Larcombe [31] was evolved into the optimal design of large civil transport aircraft. This program highlighted the significance of accuracy in the basic aircraft estimating methods.

Howe [32] described the use of the computer in the preliminary design process. He discussed what features a computer program should embody, to perform aircraft design. These features include ability of parametric variations due to iterative process and the need to extrapolate many assumptions from past experience during project design. In addition, he remarked on the limitations of computational methods, flexibility, and the difficulty of interaction techniques.

By the end of the eighties and the nineties, many powerful programs had been developed. Two separate programmes of preliminary design optimisation (GATEP) and flight-profile optimisation (SCOPE) were integrated in one program called CASTOR [33]. This program was limited to short-haul twin-turboprop aircraft. A multivariate optimisation (MVO) routine developed by Royal Aircraft Establishment was used to produce optimised aircraft designs, with a choice of three objective functions (minimum: fuel, take-off mass, and direct operating cost). The program represented a complete preliminary aircraft design methodology through its modules (e.g. geometry, mass, aerodynamics, flight performance, cost, etc.). A number of limitations were outlined by the authors, such as, the optimisation process was time consuming and the program required some degree of familiarisation from the operator which tended to discourage its use by non-experts.

ACSYNT (AirCraftSYNThesis) [34] was an interactive program, for conceptual design process. It was developed for military purposes. The first version of ACSYNT was developed in 1976 [35]. The analysis part of the program consisted of all flight performance modules. These were: Geometry, Trajectory, Aerodynamics, Propulsion, Stability, Weights, and Optimisation. The objective function of the optimiser was oriented for minimum: takeoff mass or fuel only. Optimisation times were 5 minutes on a CDC 7600 machine, while computer resources of 20-30 hours were common on the same machine. Thereafter, many modules were added to enhance the overall performance of the program. These modules included: PHIG (Programmer's Hierarchical Interactive *Graphics*) for 3D graphics, GUI for interactive user interface, TAKEOFF for detailed takeoff parameters, and ECONOMICS for evaluating the manufacturing cost, direct, and indirect costs. One drawback was that it required a significant investment in time and skills to use it effectively. *Out of a class of thirty students my experience would be that perhaps four or five would become effective users* [36].

2.3 Existing aircraft design software

Since the early nineties, several software packages have been released into the market. The Piano 1.0 [37] software was developed by Lissys Limited, which is a small UK-based company incorporated by Dr. Dimitri Simos, who has published numerous papers on aircraft design. It is continually being updated, lastest version is Piano 5.2. It is integrated software for analysing and comparing existing or projected commercial aircraft designs. It has a large database which contains more than 250 existing aircraft. The size of aircraft modelled in the database ranges from very small business jets (like the Eclipse) up to the A380 and its likely future derivatives. Its methodology is optimally tailored to FAR 25/JAR 25. The software does a complete mission analysis. It consists of many modules such as: Geometry, Weights, Aerodynamics, Performance, Cost, Parametric studies, and Optimisation. An environment emissions module has been added recently to calculate both the amount and the precise distribution of atmospheric pollutants (NOx, CO, and Hydrocarbons) emitted during a complete flight profile. The software cannot be used for detailed aircraft design. It is intended primarily to be used by airframe and engine manufacturers during their conceptual design studies.

The AAA (Advanced Aircraft Analysis) [38] software started out as a computerised version of Roskam's eight-volume textbook: *Airplane Design*, Parts I-VIII [12], featuring a friendly user interface, that performed preliminary design and analysis functions for fixed wing aircraft. Its objective was to allow students and preliminary design engineers to develop a suitable aircraft configuration with a minimum of work. The software includes many modules such as: Geometry, Weight and Balance, Performance Sizing and Analysis, Stability and Control Derivatives, and Cost Estimation. The user has the opportunity to work in Imperial units or the SI unit system. A calculator notepad is incorporated as a part of user interface, to help the user to calculate the variables and to pass the calculated results to the desired place in the program. One limitation of the software has been noticed, that it does not have mechanisms to perform parametric studies which help students to investigate the influence of changing the design variables. Other limitation is that the stability and control derivatives only deal with subsonic flow (up to about Mach = 0.7) for most derivatives.

Kroo [39] developed a system for aircraft design utilising a unique analysis architecture, graphical interface, and suite of numerical optimisation methods. The unique analysis architecture methodology is based on "loosely-coupled" systems of analysis routines, managed by executive routines [39]. Kroo was the first one who described this methodology in [40] and it has subsequently been refined and applied to several example problems. *The non-procedural architecture provides extensibility and efficiency not possible with conventional programming techniques* [39]. Program (PASS), which consists of the executive system and a collection of analysis routines, is used for analysis of subsonic transport aircraft, while the optimiser (NPSOL) is used for aircraft design studies. An expert system has been incorporated to warn the user for critical design problems through alert messages.

In 1996, Raymer developed a software package RDS [41] which implements the approach described in his book [14]. Actually, he developed two versions. The first one is for students. It consists of Aerodynamics, Weights, Propulsion, Performance, and Cost modules. 3-D CAD module has been added for design layout. The software has friendly graphical interface as well. No parametric studies module was incorporated. Output results are shown as a text and/or in a spreadsheet form. "*It has a relatively complete mission capability, and is probably the best PC program available in this price range"* [36]. The second version, which is called professional, is an expanded version with optimisation module and 3-D CAD module for design layout. It is a much more expensive version than the first one.

However, in the last ten years or so, many interactive aircraft design software have been released into the market. Madsen aircraft design program (M.A.D. V2.0.1) by Ned Madsen [42] has been developed and runs on Macintosh computers. This program was written to assist the aircraft designer with the conceptual design phase using the Mac user interface. The program is oriented toward General Aviation (GA) aircraft, and contains an excellent database of aerofoils and engines for GA aircraft. It evaluates the design in terms of weight and balance, lift and drag, stability, controllability, and performance. It does not have any fancy graphic capabilities.

Other software, which is specifically developed for light aircraft such as: GA, UAV, and helicopter, is ADS V320 Professional [43] from Optimal Aircraft Design group. It offers a convenient way to make a complete conceptual design of a single or twin engine aircraft. The cost module provides a statistical analysis which helps the user to perform the major part of the market analysis of the new aircraft. Its payload capability is limited to nine passengers only. Graphical output is limited, and the program is intended for aircraft homebuilders (54%), aircraft designers (18%), and university staff and students (24%) [43], engaged in aircraft design.

CEASIOM (Computerised Environment for Aircraft Synthesis and Integrated Optimisation Methods) [44] software is meant to support engineers in the conceptual design phase, with emphasis on improved prediction of stability and control properties achieved by higher-fidelity methods than found in the contemporary aircraft design tools. It integrates the main design disciplines, aerodynamics, structures, and flight dynamics into one application. But, it does not however carry out the entire conceptual design process. The starting point to the present CEASIOM was provided by Isikveren [45], who developed the MATLAB QCARD package for aircraft conceptual design with quasi-analytical shape definitions, aero-data correlations, and performance predictions. The main modules of CEASIOM are: Geometry (CADac) which prescribe the control surfaces in the meshing CAD model, Aerodynamic (AMB-CFD) which use VLD for low-speed aerodynamics and CFD for high-speed aerodynamics, Stability and Control (S&C) which is a static and dynamic stability and control analyser and flying-quality assessor, Aero-elastic (NeoCASS) which produce the “eta” aero-elastic coefficients and determine flutter boundaries, and Flight control system design (FCSDT) which is for flight control-law formulation, simulation, and technical decision support.

Finally, ADAS 1.0 (Aircraft Design and Analysis Software) by Nicolosi [46] is the latest software. It is "*originally intended as an educational tool, but after many improvements carried out during the past years, can be absolutely used as a research tool and it can be of some relevance also for some industrial applications*" [46]. It consists of many modules such as: preliminary estimation, wing, fuselage, aircraft's drag polar, and field and ground performance. The preliminary estimation module represents the first part of the processing. It deals with the preliminary estimation of aircraft weight (both empty and maximum takeoff), required wing area, and engine thrust/power under design requirements. Then, the user starts designing all aircraft parts. The software methodology, to estimate all aircraft characteristics, is based on classical semi-empirical laws (Roskam or ESDU) integrated with more sophisticated approaches (Multhopp or Weissinger for wing module). Although the main goal was to obtain a very user friendly platform, actually it seems very complex due to the huge number of variables per form. The software authors suggest an optimisation module is included as requirement for any future work.

2.4 Limitations of the existing software

Limitations and shortcomings of the existing software packages were mentioned in the previous section, the following limitations and shortages can be summarised here:

1. They assume prior knowledge in the field of aircraft design.
2. Evaluation of some software packages revealed that although they were designed with friendly user interfaces, their user interaction data entry forms being too complex, with a likely result of confusing the student.
3. Analysis output forms contain too much information, some of it, is not useful for the preliminary design phase (just to show the power of the software).
4. These are designed for commercial use and not for instructional use. In teaching aircraft design, the student needs to know the philosophy of the design as well

as the acceptable answer (most of them do not have the ability to perform parametric studies).

5. One of the important aspects in aircraft design is the dynamic stability. Most of these programs omit it in their synthesis process.
6. Most of them have extensive CAD capabilities, the main problem for students with CAD systems is that the aircraft design course can easily become the "learn how to use a certain CAD system" course and not to learn the philosophy, methods, and techniques of aircraft conceptual design [14].
7. They lack mechanisms to integrate with other programs for further analysis.
8. They do not have an expert system to detect and diagnose student errors.
9. From programming viewpoint, source code is not made available for modifications.

2.4.1 Prerequisites for aircraft design software for teaching

If a new package were to be developed, then the major characteristics essential for teaching purposes would be as follows:

1. Friendly graphical user interface.
2. Flexible data entry and manipulation.
3. Good visual analysis output as well as 3-view or 3D geometry.
4. Accurate analysis methods.
5. Cover all the fundamental aspects of aircraft design.
6. Fast response.
7. Parametric Studies capability.
8. Integrate easily with other programs by direct programming interface or file transfer.
9. Ability to extend to more complex analysis in smooth transition.
10. Archiving ability.
11. Availability of expert system to watch and handle user mistakes.
12. Numerical optimisation capability.
13. Generate the design configuration quickly.
14. Interaction with geometry by using design parameters.
15. Easy design maintenance (edit, modify, update any of the design parameters).

16. Software portability (machine and graphic devices independence).
17. The source code should be available for future modifications and improvements in the empirical relationships that often are required in the design process, reflecting the current practices and information.

2.5 The methodologies of teaching aircraft design

Aircraft design is a multidisciplinary subject. Teaching aircraft design requires knowledge in aerodynamics, structures, propulsion, stability and control, etc. The knowledge base is rapidly expanding and the curriculum design issues are becoming more evident. The curriculum overload results in the lecturers trying to cram too much information in the conventional “three year degree” courses, resulting in surface approach to learning. On the other hand, the present day concern of industry is that engineering students, in general, do not have the prerequisite skills to be effective practitioners. Hence, the researchers and academics have taken a note of this concern, and have evaluated methodologies that help students to become effective engineering practitioners. Two methodologies have been employed for teaching aircraft design; Problem-Based Learning and Project-Based Learning.

2.5.1 Problem-Based Learning

The original Problem-Based Learning methodology was developed for use with medical students in Canada [47]. This methodology was designed to help interns improve their diagnostic skills through working on "ill-structured problems." Medical students are introduced to a diagnostic problem, usually a patient with a complaint or illness. Using a database of information and test data about this patient and guided by a facilitator who plays the role of a coach or Socratic questioner, students are led to construct a diagnosis by generating hypotheses, collecting information relevant to their ideas (e.g., interviewing the patient, reading test data), and evaluating their hypotheses. The process, which has been used in business, architecture, law, and graduate education schools [48], combines problem statements, databases, and a tutorial process to help students hone their hypothetical-deductive thinking skills. In a similar manner, case-based methods have been used in medical, business, and legal education to help students become proficient at preparing briefs and making presentations [49].

More recently, the Problem-Based Learning methodology has been extended to mathematics, science, and social studies classes at the elementary and secondary level [50]. Gallagher, et al., [51] devised a problem-based course for high-school seniors enrolled in an Illinois school for students talented in mathematics and science. In each semester that the course was given, students were presented with two "ill-structured" problems along with raw data relevant to the problem. For example, information was presented to students about an unusually high number of persons dying of a disease with flu-like symptoms in hospitals across Illinois. Students were assigned specific tasks to (a) determine if a problem existed, (b) create an exact statement of the problem, (c) identify information needed to understand the problem, (d) identify resources to be used to gather information, (e) generate possible solutions, (f) analyse the solution using benefit /cost analysis and ripple-effect diagrams, and (g) write a policy statement supporting a preferred solution. Aside from this list of tasks, the procedure for the course was reported to be relatively non-directional. Students worked autonomously to define and seek a solution to the problem posed to them, investigating leads, asking for additional information, analysing data, etc. Results from this study focused primarily on performance, on a problem-solving test given as both a pre-test and post-test. A comparison was made between the pre-test and post-test, for the 78 students in the experimental group and a matched group that did not participate in the Problem-Based Learning course. All students were asked to describe a process for finding a solution to an ill-defined problem situation (unrelated to the problems administered in the course). Their responses were scored using a six "step" checklist. Only one of the six steps evaluated, which is the "inclusion of problem finding" showed a significant increase between the pre-test and post-test for the experimental group.

Stepien, et al. [52], in a subsequent study, described research conducted in two secondary-level settings, an elective science course for seniors and a more traditional course in American Studies for sophomores. In this study, the problem used in the science course was one designed to prompt students' consideration of ethical as well as biological issues. Likewise, the social studies problem combined historical with ethical issues. Students were asked to advise President Truman on how to bring a speedy end to the war based on an unconditional surrender by the Japanese and the assurance of a secure post-war world. The effectiveness of the two problem-solving courses was evaluated using pre-test to post-test gains on a measure of factual content for the tenth-grade course and a measure assessing the breadth of students' ethical appeals for the twelfth-grade course. In the case of the 10th-grade American Studies course, experimental students demonstrated equivalent or better knowledge of factual content as compared to a control class that studied the same period of history, but did not engage in problem solving.

As was the case in the Gallagher, et al. [51] study, students enrolled in the problem-solving course for seniors, along with a matched group of control seniors, were given an ill-structured problem as a pre-test and another such problem as a post-test. All students were instructed to outline a procedure they might use to arrive at a resolution to the problem. According to the scoring procedure employed in this study, students who took the problem-solving courses outperformed control students in the breadth of their ethical appeals and in the extent to which they tended to support their appeals with reasoned arguments.

Results on the effectiveness of a more “packaged” approach to Problem-Based Learning methodology are provided in a study by Williams, et al. [53]. In this study, conducted with 117 seventh grade science students, students taking a Problem-Based Learning programme presented via CD-ROM outperformed a control group that received more traditional instruction on a measure of knowledge of science concepts.

Additionally, there are several articles in the literature in which the authors report success for the use of a Problem-Based Learning methodology for other populations and other curriculum domains, albeit without including data. Gallagher, et al. [54] reported the successful use of the methodology with fifth-grade students on problems relating to the ecosystem; Sage [55] described the implementation of Problem-Based Learning by science and language arts teams in an elementary and a middle school; Savoie and Hughes [56] described a study involving a two-week problem-based (actually, case-based) unit for ninth-grade students focused on family dynamics; Boyce, et al. [57] described a curriculum for high-ability learners in Grades K-8, developed at the College of William and Mary, that presented highly salient, systemic problems for students to investigate (e.g., archaeology, pollution, human immunology, ecosystems);

and Shepherd [58] reported on a study conducted with fourth- and fifth-grade students using social studies problems.

This methodology of teaching aircraft design has been applied for undergraduate aeronautical engineering students in the United Kingdom for many years. It has become a widespread teaching methodology in disciplines where students have to apply knowledge not just acquire it [59]. Problem-Based Learning is a concept used to enhance multidisciplinary skills using planned project scenarios. It is an active way of learning that teaches students problem-solving skills while at the same time allowing them to acquire basic knowledge. As mentioned in the previous chapter, the students work in 5-8 student collaborative group size and learn what they need to know in order to solve a problem. The students have to work as a team to manage the design process. They are presented with problem before they have the skills and knowledge to solve it. Problem-Based Learning methodology gives the students the opportunity to choose their peers according to effort and contribution made to the group project.

2.5.2 Project-Based Learning

The methodology that organises learning around projects is called Project-Based Learning. According to the definitions found in Project-Based Learning handbooks for teachers, projects are complex tasks, based on challenging questions or problems, that involve students in design, problem-solving, decision making, or investigative activities; give students the opportunity to work relatively autonomously over extended periods of time; and culminate in realistic products or presentations [60][61]. Other defining features found in the literature include authentic content, authentic assessment, teacher facilitation but not direction, explicit educational goals, [62], cooperative learning, reflection, and incorporation of adult skills [63].

Another definition for Project-Based Learning was described by Markham [64] as: *"Project-Based Learning integrates knowing and doing. Students learn knowledge and elements of the core curriculum, but also apply what they know to solve authentic problems and produce results that matter. Project-Based Learning students take advantage of digital tools to produce high quality, collaborative products. Project-Based Learning refocuses education on the student, not the curriculum-- a shift mandated by the global world, which rewards intangible assets such as drive, passion,* *creativity, empathy, and resiliency. These cannot be taught out of a textbook, but must be activated through experience."*

An early example of applied Project-Based Learning methodology [65] is Muscatine High School, located in Muscatine, Iowa, USA. The school started the G2 (Global Generation Exponential Learning) which consists of middle and high school "Schools within Schools" that deliver the four core subject areas. At the high school level, activities may include making water purification systems, investigating service learning, or creating new bus routes. At the middle school level, activities may include researching trash statistics, documenting local history through interviews, or writing essays about a community scavenger hunt. Classes are designed to help diverse students become college and career ready after high school.

Another example [65] is Manor New Technology High School, a public high school that is part of the New Tech Network of school. Manor New Technology High School is a 100 percent project-based instruction school. Students average 60 projects a year across subjects. Since opening in fall 2007, the school has outperformed the state of Texas and Manor Independent School District in the percentage of students passing state standards in three of the four subjects tested: science, social studies, and reading/English language arts.

However, the research has been gathered on various projects that have been conducted to see the impact of Project-Based Learning methodology on students in the engineering context. One such project that aims to confront the way students learn, and the ability to integrate diversity within an academic environment, is called the Freshman Engineering project [66]. In this project, three groups of Freshman engineering students were posed with the problem of designing and constructing a stereoscopic aerial imaging platform that is low cost, light weight, and easy to control. This engineering design concept aimed to provide a hands-on approach with prominence on planning, developing and design. This project aimed to assess the experience for the students involved in terms of Project-Based Learning. The project used both engineering and aviation programmes to look at the student learning through collective projects. The outcome of this exercise was the fact, as the students realised, they could work in a team communicating and cooperating. The students also learned vital skills that could be applied to any situation such as time allocation, project management, cost constraints, and brainstorming, skills that are transferrable.

The achievements in terms of the final product, the stereoscopic aerial imaging platform varied greatly between teams. For one team, the camera activation system was successful when it came to bench testing, where as another team failed to even produce a design. To understand this difference in outcomes when it comes to Project-Based Learning methodology, the group that has cohesiveness and cohort skills is far more likely to succeed.

More recently, the Project-Based Learning methodology is practiced at the University of Hertfordshire (UH), where a selection of final year AADE students worked together to produce an Unmanned Air System (UAS) to compete in the European Students Competition on Unmanned Aircraft Systems, ESCO-UAS, in 2009 [67]. Meeting on at least a weekly basis, the team of ten-aerospace students, each of them have their own specialist area of expertise, worked together to design and manufacture the UAS. To realise this project as a form of Project-Based Learning methodology, the essential activities represent some degree of difficulty to the student that can’t be carried out with information that has already been learnt. From this point of view, it is obvious that the UAS project is a type of Project-Based Learning, as students are continuously learning through the course of the design and manufacture exercise.

Although only in the fourth year of running the project, it is evident that students are gaining deeper understanding and knowledge not only in a text-book type manner but actually getting hands on experience and developing their interpersonal skills. Moreover, this style of learning is a type of long-life learning, as students are actually putting their knowledge into practice. The UAS project allows the exploration of the imagination within the team members in terms of design ideas and actually achieving the desired end result. The teams have showed dedication, the willingness to work together, and the ingenuity to succeed. Many problems have been put in front of the team, but through shear-determination, the inconveniences have been overcome. The UAS project at UH is a prime example of how Project-Based Learning methodology can enthuse students to excel in their academic studies [67].

From the above examples, Project-Based Learning methodology emphasises learning activities that are long-term, interdisciplinary and student-centred. Unlike traditional, teacher-led classroom activities, students often must organise their own work and manage their own time in a project-based class. Project-based instruction differs from traditional inquiry by its emphasis on students' collaborative or individual artefact construction to represent what is being learned [68].

The core idea of Project-Based Learning methodology is that real-world problems capture students' interest and provoke serious thinking as the students acquire and apply new knowledge in a problem-solving context. The lecturer (or teacher) plays the role of facilitator, working with students to frame worthwhile questions, structuring meaningful tasks, coaching both knowledge development and social skills, and carefully assessing what students have learned from the experience [69].

The main criticism of Project-Based Learning methodology is that it may be inappropriate in mathematics, the reason being that mathematics is primarily skill-based at the elementary level. Transforming the curriculum into an over-reaching project or series of projects does not allow for the necessary practice at particular mathematical skills. For instance, factoring quadratic equations in elementary algebra is something that requires extensive practice [70].

On the other hand, a teacher could integrate a Project-Based Learning methodology into the standard curriculum, helping the students see some broader contexts where abstract quadratic equations may apply. For example, Newton's law implies that tossed objects follow a parabolic path, and the roots of the corresponding equation correspond to the starting and ending locations of the object. Another criticism of Project-Based Learning methodology is that measures that are stated as reasons for its success are not measurable using standard measurement tools, and rely on subjective rubrics for assessing results [71].

It should be noted, not to confuse between Problem-Based Learning and Project-Based Learning. Both share much in common. Both share the same abbreviation (PBL). The differences include [72]:

1. *Project tasks are closer to professional reality and therefore take a longer period of time than problem-based learning problems.*
2. *Project work is more directed to the application of knowledge, whereas problem-based learning is more directed to the acquisition of knowledge.*

3. *Project-based learning is usually accompanied by subject courses (e.g. maths, physics etc. in engineering), whereas problem-based learning is not.*
4. *Management of time and resources by the students as well as task and role differentiation is very important in project-based learning.*
5. *Self-direction is stronger in project work, compared with problem-based learning, since the learning process is less directed by the problem.*

Finally, the research on the application of technology to learning and instruction has led, in general, to an interest in using technology as a "cognitive tool" as well as extension of student capabilities. In addition, technology has, among its touted benefits, the value of making the knowledge construction process explicit, thereby helping learners to become aware of that process [73]. *"Using technology in project-based science makes the environment more authentic to students, because the computer provides access to data and information, expands interaction and collaboration with others via networks, promotes laboratory investigation, and emulates tools experts use to produce artefacts."* [74]. Hence, this research (iADS) is a practical example of incorporating computer software into both Problem-Based Learning and Project-Based Learning methodologies.

2.6 Methods of aircraft design formulae

Designing an aircraft is based on many formulae suggested by many analysts. Aircraft design formulae have different accuracy based on the theories used and the assumptions made in the development of their formulae. The preliminary formulae can be classified into three categories: Empirical, Analytical, and Semi-analytical.

2.6.1 Empirical methods

Empirical methods are the earliest and most frequently used [75]. They are based on the use of historical data of the manufactured aircraft to develop empirical relationships in order to predict, for example, the weight of a new aircraft, using simple regression techniques. To achieve a successful application of this method for the design under consideration, the aircraft weight being estimated must be similar in terms of configuration and characteristics to those of the reference database [76]. This method can be divided into two categories; fixed fraction, and statistical equations methods. The fixed fraction method is the simplest form of the empirical method due to its elementary nature. The aircraft breaks down into its components and sub-components. Each component (or sub-component) is calculated as a fraction of the whole aircraft (or whole component). For example, wing drag is calculated as a fraction of aircraft drag, stud weight is calculated as a fraction of wing weight, and so on. The statistical equations method is the most widely used of the empirical methods. It is based on correlating historical data by means of statistical equations known as the Power Law formula, [77]. For example, the weight of the component of interest is obtained by summing up all the physical characteristics combinations, which influences the weight of the component [76]. The empirical coefficients are determined by either a least square curve fitting process or by constrained regression analysis techniques, using weights and geometric data of actual components. The constrained regression analysis technique plots data on a logarithmic scale as a straight line. All available data can be plotted against the correlation parameter. A regression analysis produces a trend line through the data, and the coefficients and exponent can be determined from the intercept and slope of the line respectively, [76]. The trend line can be altered for different combinations of design parameters, thus changing the value of the coefficients to suit different types of aircraft. The selection of the type of design parameter is of vital importance for any success in the application of this method and has to consist of one or more characteristics of the component or the vehicle, [77].

The empirical methods are the most generally used weight estimation techniques due to their simplistic nature [78]. These methods can be used to generate fast and accurate estimations for different configurations of aircraft [79][80][81]. On the other hand, empirical methods have several limitations. First of all, its use of historical data of the existing aircraft, which are very similar in both shape and size. Secondly, they do not possess the ability to account for technological advancements due to its elementary nature. Thirdly, the accuracy of the calculations depends on the size and nature of the referenced database. Finally, the selection of the correlation parameters does not necessarily guarantee satisfactory results [82].

2.6.2 Analytical methods

The analytical methods offer exact techniques and are related to individual aspects of individual systems and not to the aircraft as a whole. These methods require prior knowledge in the field of aircraft system design and allow inclusion of weight control measures [81]. Analytical methods have three fundamental features; design intent, size criteria, and production design. Once these features are established, the computation process takes place by identifying and analysing the loads that drive the design of the particular component, taking into account any physical constraints and limits [83]. The main benefit of these methods is the ability to incorporate new technologies, materials, and concepts due to the detail and load analysis component [81]. Also, the analytical methods tend to be more accurate than the empirical methods and encourage multidisciplinary design, in that engineers from various departments are involved in the design from the beginning. On the other hand, the analytical methods have limitations. The major limitation of these methods is that they are extremely detailed which require vast amount of information, which may not be available in the early phases of the design [84][85]. Therefore, these methods tend to be used in detailed design phase rather than in preliminary design phase. Other limitation is that it is very hard to incorporate risk assessments in the design procedures, thereby making them risky because the analytical methods are more oriented toward groups and components [81].

2.6.3 Semi-analytical methods

The semi-analytical methods combine the benefits of both analytical and empirical methods. They are so-called (rather than semi-empirical) because the predictions are based largely on analytically derived estimates which are correlated to reported data of the existing aircraft. Some of the individual penalty and special feature estimates may be statistically based [81][85]. A good example of an application of these methods is the wing structure weight estimation proposed by Torenbeek [10]. It starts with estimation of the wing structural box weight using station cut analysis, an analytical technique used to determine the shear, torsion and optimum bending stresses. The total wing weight is then determined by applying a series of analytical and statistically based increments to the estimated weight to account for non-optimum factors such as fasteners, cut-outs, wing fold, splices and joints, fuselage attachments, fuel containment etc. Lastly for the control surfaces, the fixed trailing edge weights can be estimated separately using statistical methods [81][86]. The main benefit of the semi-analytical methods is their versatility. These methods have the highest accuracy than the others and require less data compared to analytical methods [87]. Other benefit is that the semi-analytical methods are able to incorporate new technologies and concepts due to their analytical component. Also, these methods offer continuity and accountability for details and changes through the latter stages of the design [81][85]. The main limitation of these methods is that the parametric components associated with these methods, make them reliant on historical data, which limit their applicability to unconventional aircraft configurations that fall outside the scope of their reference database [10]. Torenbeek's approach [10] is based on these methods and therefore, his formulae set are employed in the development of the iADS software.

2.7 Conclusions

The use of computers in higher education to aid learning is widespread. The integration of computer software into Problem-Based Learning can encourage students to learn effectively and in depth. The power of software can be harnessed to make the subject material much more interesting than before, and by varying the ways in which information is delivered. After SYNAC II and CAAD software packages for aircraft design, several software packages have been released into the market. PIANO, AAA, and RDS are examples of such packages. One drawback of these packages is that they assume prior knowledge in the field of aircraft design. Also, their main purpose being the preliminary aircraft design in a commercial environment, and are not intended for instructional use. For instructional use the key features for aircraft design software were outlined. Teaching aircraft design is based on two methodologies; Problem-Based Learning and Project-based Learning. These methodologies and the differences between them were indicated. Implementation of any aircraft design software is based on different formula sets suggested by various analysts, and the accuracy of the predictions is largely based on the methods employed. These formula sets are classified into three categories; empirical, analytical, and semi-analytical methods. Benefits and limitations of each category were outlined.

Chapter THREE

Preliminary Aircraft Design Process

3.1 Introduction

The aircraft design process consists of three phases as mentioned on the first chapter. In the conceptual phase, where the design process is based on concepts without resorting to precise calculations, all parameters are determined by decision making processes and selection techniques. An estimation of maximum takeoff weight (MTOW) is the first step in this phase. Zero-order sizing methodology, which is found in most textbooks, is the general technique used for that estimation. MTOW is divided into four sub-weights, which are: empty, crew, payload, and fuel weight. Crew and payload weights are evaluated from requirements whereas empty weight is estimated from statistics. Fuel weight is calculated using a modified Brequet range equation. The result of this process is an inaccuracy of 10-15%. All the output design parameters of this phase become preliminary input variables. Analytical, empirical and semi-empirical equations are used in the second design phase. The mass estimation is improved and inaccuracies of 5-10% remain, and occasionally as low as 5%. As accuracy of mass estimation is increased, the complexity of calculation increases, and hence the number of design variables needed for a more refined estimate increases. Design of an aircraft requires several hundreds of variables to be manipulated. Some of them are empirically estimated, while others are analytically computed. To understand the aspects of preliminary design process, first input variables required to perform the process are examined.

3.2 Preliminary input variables

Input variables comprise of two categories: Aircraft geometry variables and Mission requirements variables.

3.2.1 Aircraft geometry input variables

To facilitate exploring these variables, the aircraft variables are grouped into four sections, which are: Wing, Fuselage, Tail, and Propulsion.

3.2.1.1 Wing section

The wing section is the most important component of the aircraft. Its primary function is to generate lift. From wing geometry view point, the following design variables are considered:

1. **Reference Area (S_w):** This is the most important wing parameter. It is the trapezoidal portion of the wing projected into the centreline. It is evaluated when the wing loading is selected. As the wing area is increased, both lift produced and wing weight, are proportionally increased.
2. **Aspect Ratio (A_w):** An important key characteristic that defines the shape of the wing is the aspect ratio. It is the ratio of the wing length to its chord. High aspect ratio indicates a long, narrow wing, while low aspect ratio indicates a short, wide wing.
3. **Taper Ratio (λ_w):** It defines the shape of the wing. It is the ratio between the tip chord and the centreline chord. It affects the distribution of the lift along the span of the wing.
4. **Sweep Angle (Λ):** The major use of a swept wing is to reduce the adverse effects of transonic and supersonic flow. Lateral stability is improved by sweeping the wing.
5. **Thickness Ratio (t_{c_w}):** This is the ratio of the maximum thickness of the aerofoil to the length of its chord. It provides the major influence on the profile drag. It has direct effect on drag, maximum lift, stall characteristics, and wing weight.
6. **Wing root thickness to tip Thickness Ratio (w_{TCRTR}):** It describes the relationship between the root and tip thickness. Hence, it has great effect on the pressure distribution on the wing.
7. **Wing configuration:** This is the location or positioning of the wing with respect to the fuselage. Three locations are categorised: high, mid, and low. Each configuration has its own benefits and disadvantages. Due to the structural carry-through problem, transport aircraft have either high or low wing configuration.
8. **Incidence Angle (w_{inc}):** This is the angle at which the wing is installed on the fuselage, measured relative to the axis of the fuselage.

From an aerodynamics view point, aerofoil section is important. The correct choice ensures the generation of the optimum pressure distribution on both upper and lower surfaces of the wing such that the required lift is created with the lowest drag. The following design parameters, which are related to wing aerodynamics, represent the key characteristics of the aerofoil section:

1. **Lift Curve Slope (C_{L_w}):** This is the slope of the variation of lift coefficient with respect to the change in the angle of attack. The better aerofoil is one with a higher curve slope.
2. **Maximum Lift Coefficient ($C_{L_{max}}$):** This is the maximum capacity of an aerofoil to produce non-dimensional lift. The higher maximum lift coefficient, the lower the stall speed (i.e. safer flight).
3. **Ideal Lift Coefficient (C_{L_d}):** This is the lift coefficient at which the drag coefficient does not vary significantly with the slight variations of angle of attack. It usually corresponds to the minimum drag coefficient.
4. **Pitching Moment Coefficient (C_{m_w}):** Pitching moment is the moment (or torque) produced by the aerodynamic force on the aerofoil if that aerodynamic force is considered to be applied, not at the centre of pressure, but at the aerodynamic centre of the aerofoil. The pitching moment on the wing of an aeroplane is part of the total moment that must be balanced using the lift on the horizontal stabiliser. Pitching moment coefficient is defined as a result of the pitching moment divided by the product of dynamic pressure, wing area, and the chord of the aerofoil [88]. Pitching moment is, by convention, considered to be positive when it acts to pitch the aerofoil in the nose-up direction. Conventional cambered aerofoils supported at the aerodynamic centre pitch nose-down so the pitching moment coefficient of these aerofoils is negative [89].

High lift devices are moving surfaces intended to increase lift during certain flight's conditions. The most common type of high lift devices are flaps which are usually located at the trailing edge of the wing and used to increase the wing camber to help produce more lift. The following design variables that related to flaps should be assigned:

1. **Flap outboard span to wing span ratio ($R_{b_{f_o}/b}$):** Flap span depends on the amount required for the ailerons. Transport aircraft have a large flap span (more than 70% of the wing span) and small inboard ailerons (less than 30% of wing span) used for gentle manoeuvre at high speeds.
2. **Flap chord to wing chord ratio (R_{c_f/c_w}):** Since the flap deflection increases the wing drag, due to the increase in the frontal area, the flap chord must not to be too high that it eliminates its advantages.
3. **Maximum flap deflection at take-off (δ_{fto}):** This is an important parameter to achieve the required lift increment at the takeoff stage.
4. **Maximum flap deflection at landing (δ_{flan}):** This is an important parameter to achieve the required drag increment at the landing stage.
5. **Type of flap:** Four types are included which are single-slotted, double-slotted, single-slotted fowler, and double-slotted fowler flaps. As the flap complexity increases, weight, efficiency, and cost increases.

3.2.1.2 Fuselage section

This is the main body section that holds crew and passengers and/or cargo. The shape of the fuselage is normally determined by the mission requirements, as specified by the top level requirements document. Fighter aircraft have a very slender fuselage, whilst transport aircraft have a wider fuselage to carry the passengers. The cylindrical, elliptical or double bubble fuselage sections are terminated at the front and rear by a cone. These are generally referred to as the nose and tail cone sections. Fuselage configuration has major effects on the cost of flight operation, cost of aircraft, performance, passenger comfort, and aircraft life. From the point of view of fuselage geometry, the following fuselage design variables, which are evaluated in the conceptual design process depending on seat configuration, seat pitch, and number of crew, are required for the preliminary design process:

1. **Fuselage Diameter (D_{fus}).**
2. **Nose Cone Length ($l_{1_{fus}}$).**
3. **Cabin Length ($l_{2_{fus}}$).**

4. **Tail Cone to Fuselage Diameter Ratio ($R_{l_{3_{fus}}/D_{fus}}$).**
5. **Wing Position to Fuselage Length Ratio ($R_{p_w/l_{fus}}$).**
6. **Number of Seats per Row ($N_{seat/row}$).**

3.2.1.3 Tail section

A tail is a little wing. The main difference between a wing and a tail is that a wing is used to provide lift whilst a tail is used to provide trim, stability, and control. Trim refers to generating a moment about the centre of gravity to balance some other moment produced by the aircraft. Stability refers to restoration of the aircraft from an upset in pitch, yaw or roll, and the ability of the aircraft to adopt a new stable attitude or to return to its initial state. Control refers to the provision of adequate control power at all critical conditions such as nose-wheel lift-off, low-speed flight with flaps deployed, engine-out flight at low speeds, and spin recovery. The tail section consists of two parts, the horizontal tail (also called tail-plane) and vertical tail (also called fin). Both have similar design variables to the wing. These are:

1. **Aspect Ratio (A_{ht}, A_{vt}).**
2. **Taper Ratio ($\lambda_{ht}, \lambda_{vt}$).**
3. **Thickness Ratio ($t_{c_{ht}}, t_{c_{vt}}$).**
4. **Sweepback Angle (Λ_h, Λ_v).**
5. **Volume coefficient (V_{ht}, V_{vt}):** Instead of reference area, this coefficient is used to calculate the tail area.
6. **Tail configuration:** Transport aircraft have either a conventional type or T-type. Military aircraft may have more than one fin for stability requirements, or a novel tail configuration.

3.2.1.4 Propulsion section

This consists of two parts: propulsion system (engine) and nacelle system. For engines, two objectives must be achieved. Thrust must balance the drag in cruise, and thrust must exceed the drag in order to accelerate, during climb. The propulsion section has a direct effect on performance, aircraft cost, flight time, and aircraft life. From an engine view point, the following variables are considered:

1. **Engine Scale Factor ($N_{sf_{eng}}$):** Full ground and altitude data for a Rolls-Royce –Allison engine were available. Also, an equation that allowed computation of reference thrust as a function of Mach number, altitude, temperature and density was obtained after multi-variable regression. The reference thrust for this engine is 6797 (lbs). It is a high by-pass ratio engine and representative of modern day gas turbine technology. A scale factor is used to up size or down size the engine to produce the maximum thrust required.
2. **Engine Length (l_{eng}).**
3. **Engine Span-wise Distance ($x_{l_{eng}}$):** The horizontal distance from the fuselage centreline to its location.
4. **Number of Engines (N_{eng}).**
5. **Engine Thrust (T_{eng}).**

The nacelle system is the aerodynamic structure that surrounds a jet engine. It includes not only the parts commonly referred to as engine cowling, but also other parts such as: inlet cowl, fan cowl, thrust reverser, core cowl, and nozzle. The design variables related to a nacelle system are:

1. **Nacelle Diameter (D_{nac}).**
2. **Nacelle Length (l_{nac}).**
3. **Distance of Nacelle Leading Edge to Forward of Wing Quarter-Chord Line ($l_{nac_{le_{\bar{c}/4}}}$).**

3.2.2 Mission requirements variables

The second category of input variables, which are based on customer requirements, aircraft configuration, and power plant selection, are very important in analysing the flight profile performance, they are:

1. **Main Mission Climb Speed (V_{clb}).**
2. **Main Mission Cruise Speed (V_{cru}).**
3. **Main Mission Descent Speed (V_{dsc}).**
4. **Diversion Cruise Speed (V_{Dvcru}).**
5. **Design Dive Speed (V_{dive}).**

6. **Main Stage Length (s_l).**
7. **Diversion Stage Length (s_{Dv}).**
8. **Main Mission Cruise Height (H_{cru}).**
9. **Diversion Cruise Height ($H_{cru_{Dv}}$).**

3.3 Preliminary process synthesis

As more detailed information is available, the design process becomes more accurate. Hence, the preliminary design process yields accurate results when compared to the conceptual process. Many formula sets have been suggested by many analysts. These formulae have different accuracy based on the theories used and the assumptions made in the development of their formulae. The preliminary formulae can be classified into three categories: Empirical, Analytical, and Semi-analytical. Each category has its benefits and limitations, as mentioned in the previous chapter. Torenbeek's approach [10] is an example of semi-analytical category and his formulae set are employed in the development of the iADS software. This choice is based on the methodology yields accurate results. And because of inherent accuracy, in the sphere of aircraft design, his formulae set are widely used in the preliminary design process. The classical synthesis approach can be considered as separate inter-related sections. These sections are: **Geometry, Weights, CG, Stability, Aerodynamics, Performance,** and **Cost**. Presented in the subsequent sections are the relationships, given in Torenbeek [10], for the purposes of completeness. Any deviations from his formulations will be duly noted and outlined.

3.3.1 Geometry

Aircraft Geometry allows other design variables to be computed, which in turn are to be used in the subsequent detailed analysis. Starting with the wing component, the following output parameters are:

1. Wing Span:-

$$b_w = \sqrt{A_w \times S_w} \tag{3.1}$$

2. Centreline Chord:-

$$c_{r_w} = \frac{2\times S_w}{b_w\times(1+\lambda_w)} \tag{3.2}$$

3. Tip Chord:-

$$c_{t_w} = c_{r_w} \times \lambda_w \tag{3.3}$$

4. Mean Geometric Chord:-

$$\bar{c}_w = \frac{c_{r_w}+c_{t_w}}{2} \tag{3.4}$$

5. Mean Aerodynamic Chord:-

$$\bar{c} = \frac{2}{3} \times c_{r_w} \times \left(\frac{1+{\lambda_w}^2}{1+\lambda_w}\right) \tag{3.5}$$

6. Root Thickness:-

$$w_{TCRT} = w_{TCRTR} \times t_{c_w} \tag{3.6}$$

7. Exposed Wing Span:-

$$b_{x_w} = b_w - D_{fus} \tag{3.7}$$

8. Exposed Wing Area:-

$$S_{x_w} = S_w - \left(c_{r_w} \times D_{fus}\right) \tag{3.8}$$

9. Exposed Wing Aspect Ratio:-

$$A_{x_w} = \frac{{b_{x_w}}^2}{S_{x_w}} \tag{3.9}$$

10. Distance from Fuselage Nose to Wing Quarter Root:-

$$l_{fqc} = l_{p_w} + \frac{1}{4} \times c_{r_w} \tag{3.10}$$

11. Distance from Fuselage Nose to Wing Centre of Pressure:-

$$l_{fac} = l_{fqc} + \frac{b_w}{4} \times tan\Lambda \tag{3.11}$$

For Flap section, these parameters are:

1. Flap Span Inboard Position:-

$$b_{f_i} = D_{fus} \tag{3.12}$$

2. Flap Span Outboard Position:-

$$b_{f_o} = R_{b_{f_o}/b} \times b_w \tag{3.13}$$

3. Flap Inboard Span to Wing Span Ratio:-

$$R_{b_{f_i}/b} = \frac{b_{f_i}}{b_w} \tag{3.14}$$

4. Flapped Wing Area:-

$$S_{w_{flap}} = S_w \times \frac{\left(b_{f_o} - b_{f_i}\right)}{b_w} \times \left(1 + \frac{(1-\lambda_w)}{(1+\lambda_w)}\right) \times \left(1 - \frac{\left(R_{b_{f_o}/b} - R_{b_{f_i}/b}\right)}{b_w}\right) \tag{3.15}$$

5. Flap area:-

$$S_f = S_{w_{flap}} \times R_{c_f/c_w} \tag{3.16}$$

For Fuselage section:

1. Tail Cone Length:-

$$l_{3_{fus}} = R_{l_{3_{fus}}/D_{fus}} \times D_{fus} \tag{3.17}$$

2. Fuselage Length:-

$$l_{fus} = l_{1_{fus}} + l_{2_{fus}} + l_{3_{fus}} \tag{3.18}$$

3. Fuselage Wetted Area:-

$$S_{w_{fus}} = \pi \times D_{fus} \times \left(l_{1_{fus}} \times k_{w1} + l_{2_{fus}} + l_{3_{fus}} \times k_{w3}\right) \tag{3.19}$$

Where: k_{w1} & k_{w3} are shape parameters. For transport aircraft, $k_{w1} = k_{w3} = 0.75$.

For Vertical tail section:

1. Tail Arm in case of Rear Engine (Fuselage Mounted):-

$$l_{vt} = l_{fus} - l_{fac} \tag{3.20}$$

2. Tail Arm in case of Engine (Wing Mounted):-

$$l_{vt} = l_{fus} - l_{fac} - 0.75 \times c_{r_{vt}} + \frac{b_{vt}}{2} \times tan\Lambda_v \tag{3.21}$$

3. Area of the vertical tail:-

$$S_{vt} = \frac{S_W \times b_W \times V_{vt}}{l_{vt}} \tag{3.22}$$

4. Root Chord:-

$$c_{r_{vt}} = 2 \times \frac{\sqrt{S_{vt}/A_{vt}}}{(1+\lambda_{vt})} \tag{3.23}$$

5. Span:-

$$b_{vt} = \sqrt{S_{vt} \times A_{vt}} \tag{3.24}$$

6. Mean Geometric Chord:-

$$\bar{c}_{vt} = \frac{S_{vt}}{b_{vt}} \tag{3.25}$$

For horizontal Tail section:

1. Tail Arm for Conventional Configuration:-

$$l_{ht} = l_{vt} + \frac{b_{ht}}{4} \times tan\Lambda \tag{3.26}$$

2. Tail Arm for T-Tail Configuration:-

$$l_{ht} = l_{vt} + \frac{b_{vt}}{2} \times tan\Lambda_v + \frac{b_{ht}}{4} \times tan\Lambda \tag{3.27}$$

3. Area:-

$$S_{ht} = S_w \times \bar{c} \times \frac{V_{ht}}{l_{ht}} \tag{3.28}$$

4. Root Chord:-

$$c_{r_{ht}} = 2 \times \frac{\sqrt{S_{ht}/A_{ht}}}{(1+\lambda_{ht})} \tag{3.29}$$

5. Span:-

$$b_{ht} = \sqrt{S_{ht} \times A_{ht}} \tag{3.30}$$

6. Mean Geometric Chord:-

$$\bar{c}_{ht} = \frac{S_{ht}}{b_{ht}} \tag{3.31}$$

For Nacelle section, Nacelle Wetted Area is:-

$$S_{w_{nac}} = \pi \times D_{nac} \times l_{nac} \times \left(0.5 + 0.135 \times \frac{l_{nac_{le_{\bar{c}/4}}}}{l_{nac}}\right)^{0.667} \times \left(1.015 + \frac{0.3}{(l_{nac}/D_{nac})^{1.5}}\right) \tag{3.32}$$

3.3.2 Aircraft weight

Weight for each aircraft component is calculated using build-up methodology that allows the maximum takeoff weight to be broken down into its components and adds them to determine empty, operational empty, zero-fuel, and maximum takeoff weights. In order to calculate component weights, pre-calculations for the load factors (limit and ultimate) are required as follows:

Initially, the limit load factor which is the greater of the gust and manoeuvre factors is evaluated. These load factors are determined in accordance with airworthiness requirements [7][8]. The following relationships [13] are used: -

$$N_{gust} = \frac{1+6.3\times A_w \times S_w \times V_{dive}}{m_{to}(2+A_w)} \tag{3.33}$$

$$N_{manu} = 2.1 + \frac{10900}{4530+m_{to}} \tag{3.34}$$

$$N_{manu} = 2.5 \tag{3.35}$$

$N_{manu} = the\ greatest\ of\ Equations\ (3.34)\ and\ (3.35).$

The second step is to calculate the ultimate load factors of both gust and manoeuvre:-

$$(N_{gust})_{ult} = 1.5 \times N_{gust} \tag{3.36}$$

$$(N_{manu})_{ult} = 1.65 \times N_{manu} \tag{3.37}$$

$N_{ult} = the\ greatest\ of\ Equations\ (3.36) and\ (3.37).$

The Individual aircraft components are determined as follows:

1. Wing mass:-

$$m_w = 0.00667 \times N_{ult}^{0.55} \left(t_{c_w} \times c_{r_w}\right)^{-0.3} \times \left(\frac{b_w}{cos\Lambda_{\frac{1}{2}}}\right)^{1.05} \times \left(1 + \sqrt{\frac{1.905 \times cos\Lambda}{b_w}}\right) \times \left(\frac{m_{zf}}{S_w}\right)^{-0.3} \times m_{zf} \tag{3.38}$$

2. Fuselage mass:-

$$m_{fus} = 0.23 \times S_{w_{fus}}^{1.2} \times \sqrt{\frac{V_{dive} \times l_{ht}}{2 \times D_{fus}}} \tag{3.39}$$

3. Tail mass:-

$$m_T = 0.051 \times \frac{V_{dive} \times (S_{ht} + S_{vt})^{1.2}}{\sqrt{cos\Lambda}} \tag{3.40}$$

4. Propulsion mass:-

$$m_{pro} = 1.377 \times m_{eng} \times N_{eng} + 0.055 \times T_{eng} \times N_{eng} \tag{3.41}$$

5. Undercarriage mass:-

$$m_{uc} = 0.04 \times m_{to} \quad , \ for\ low\ wing \tag{3.42}$$

$$m_{uc} = 0.045 \times m_{to} \quad , \ for\ high\ wing \tag{3.43}$$

6. Surface Controls mass:-

$m_{sur} = 0.4915 \times m_{to}^{2/3}$ (3.44)

7. Systems mass:- For accurate estimation, further broken down to its sub-components as follows:

   a. APU

   $m_{apu} = 2.2 \times m_{apu_{dry}}$ (3.45)

   b. Instruments and Avionics

   $m_{ins} = 0.347 \times \left(\frac{m_{to}}{2}\right)^{0.555} \times \left(\frac{s_l}{1000}\right)^{0.25}$ (3.46)

   c. Hydraulics and pneumatics

   $m_{hyd} = 0.015 \times \left(\frac{m_{to}}{2}\right) + 272$ (3.47)

   d. Electrical

   $m_{ele} = 10.8 \times \left(\pi \times l_{2_{fus}} \times \left(0.9 \times D_{fus}\right)^2\right)^{0.7} \times \left(1 - 0.18 \times \left(\pi \times D_{fus} \times \left(0.9 \times D_{fus}\right)^2\right)^{0.7}\right)$ (3.48)

   e. Air conditioning and Anti-icing

   $m_{acic} = 14 \times l_{2_{fus}}^{1.28}$ (3.49)

Adding the sub-components defined by Equations (3.45) to (3.49), the total systems mass:

$m_{sys} = m_{apu} + m_{ins} + m_{hyd} + m_{ele} + m_{acic}$ (3.50)

8. Furnishings mass:-

$m_{furn} = 0.196 \times m_{zf}^{0.91}$ (3.51)

Hence, Empty weight using Equations (3.38) to (3.44), (3.50) and (3.51) is:

$$m_e = m_w + m_{fus} + m_T + m_{uc} + m_{pro} + m_{sur} + m_{sys} + m_{furn} \tag{3.52}$$

9. Operating items mass:-

$$m_{op\_it} = 8.617 \times N_{pas} \quad , for\ short\ range \tag{3.53}$$

$$m_{op\_it} = 14.97 \times N_{pas} \quad , for\ long\ range \tag{3.54}$$

10. Flight crew mass:-

$$m_{fl_{crew}} = 93 \times N_{fl_{crew}} \tag{3.55}$$

11. Flight attendants mass:-

$$m_{fl_{att}} = 68 \times N_{fl_{att}} \tag{3.56}$$

Operating empty weight using Equations (3.52) to (3.56) is defined as:

$$m_{oe} = m_e + m_{op\_it} + m_{fl_{crew}} + m_{fl_{att}} \tag{3.57}$$

12. Payload mass:- Assuming 90.72 kg (200 lbs) per passenger, the payload mass is:

$$m_{pay} = 90.72 \times N_{pas} \tag{3.58}$$

Zero fuel weight is then defined as the sum of Equations (3.57) and (3.58):

$$m_{zf} = m_{oe} + m_{pay} \tag{3.59}$$

13. Fuel mass m_{fuel} is an input variable, and varies depending on the mission stage requirements. From a design standpoint, this figure normally is the maximum fuel that can be carried by the aeroplane, so that the maximum loaded weight of the aircraft can be known. This figure is important for the majority of performance calculations.

Maximum takeoff weight is therefore the sum of Equation (3.59) and m_{fuel}:

$$m_{to} = m_{zf} + m_{fuel} \tag{3.60}$$

3.3.3 Centre of gravity (CG)

Aircraft must be designed to achieve good stability and control properties and adequate flexibility in loading conditions. For this reason, computation of the CG is required to be determined for several loading cases. Loading cases include variable payload and fuel. Engine and tail configurations are taken into consideration as well. Although arbitrary cases can be used, preference is to represent common current practice.

Initially, the moment of an empty aircraft is determined by calculating the moment of each component around a reference point (Datum) and summing them together as in the following:

1. Wing moment:-

$$M_w = \left(l_{f_{qc}} + 0.25 \times \bar{c} + \frac{b_w}{6} \times tan\Lambda\right) \times m_w \tag{3.61}$$

2. Fuselage moment:-

Let:

$$x = l_{1_{fus}} \times k_{w1} + l_{2_{fus}} + l_{3_{fus}} \times k_{w3} \tag{3.62}$$

Then:

$$M_{fus} = \left(0.561 \times l_{1_{fus}}\right) \times \left(\frac{l_{1_{fus}} \times k_{w1} \times m_{fus}}{x}\right) + \left(l_{2_{fus}} + \frac{l_{2_{fus}}}{2}\right) \times \left(\frac{l_{2_{fus}} \times m_{fus}}{x}\right) + \left(l_{1_{fus}} + l_{2_{fus}} + 0.413 \times l_{3_{fus}}\right) \times \left(\frac{l_{3_{fus}} \times k_{w3} \times m_{fus}}{x}\right) \tag{3.63}$$

3. Horizontal tail moment:-

$$M_{ht} = \left(l_{fac} + l_{ht} + 0.25 \times \bar{c}_{ht}\right) \times \left(\frac{S_{ht}}{S_{ht}+S_{vt}}\right) \times m_T \quad , for\ low\ tail \tag{3.64}$$

$$M_{ht} = \left(l_{fac} + l_{ht}\right) \times \left(\frac{S_{ht}}{S_{ht}+S_{vt}}\right) \times m_T \quad , for\ high\ tail \tag{3.65}$$

4. Vertical tail moment:-

$$M_{vt} = \left(l_{fac} + l_{vt} + 0.25 \times \bar{c}_{vt}\right) \times \left(\frac{S_{vt}}{S_{ht}+S_{vt}}\right) \times m_T \quad , for\ low\ tail \tag{3.66}$$

$$M_{vt} = \left(l_{fac} + l_{vt}\right) \times \left(\frac{S_{vt}}{S_{ht}+S_{vt}}\right) \times m_T \quad , for\ high\ tail \tag{3.67}$$

5. Nacelles moment:-

$$M_{nac} = \left(l_{fqc} + 0.5 \times l_{nac} + x_{l_{eng}} \times tan\Lambda - l_{nac_{le_{\bar{c}/4}}}\right) \times \left(0.055 \times T_{eng} \times N_{eng}\right) \quad , for\ wing\ mounted \tag{3.68}$$

$$M_{nac} = \left(l_{1_{fus}} + l_{2_{fus}} + 0.5 \times l_{nac}\right) \times \left(0.055 \times T_{eng} \times N_{eng}\right) \quad , for\ fuselage\ (rear)\ mounted \tag{3.69}$$

6. Engines moment:-

$$M_{eng} = \left(l_{fqc} + 0.25 \times l_{nac} + x_{l_{eng}} \times tan\Lambda - l_{nac_{le_{\bar{c}/4}}}\right) \times \left(1.377 \times m_{eng} \times N_{eng}\right) \quad , for\ wing\ mounted \tag{3.70}$$

$$M_{eng} = \left(\left(l_{1_{fus}} + l_{2_{fus}} + 0.25 \times l_{nac}\right)\right) \times \left(1.377 \times m_{eng} \times N_{eng}\right) \quad , for\ fuslage\ (rear)\ mounted \tag{3.71}$$

7. Undercarriage moment:-

$$M_{uc} = \left(l_{fqc} + 0.5 \times \bar{c} + \frac{b_w \times tan\Lambda}{6}\right) \times m_{uc} \tag{3.72}$$

In the absence of detailed data for surface controls, systems, and furnishings at this stage, it is assumed that the CG positions for them are in the centre of the cabin section, i.e.:-

8. Surface Controls moment:-

$$M_{sur} = \left(l_{1_{fus}} + 0.5 \times l_{2_{fus}}\right) \times m_{sur} \tag{3.73}$$

9. Systems moment:-

$$M_{sys} = \left(l_{1_{fus}} + 0.5 \times l_{2_{fus}}\right) \times m_{sys} \tag{3.74}$$

10. Furnishings moment:-

$$M_{furn} = \left(l_{1_{fus}} + 0.5 \times l_{2_{fus}}\right) \times m_{furn} \tag{3.75}$$

Total empty aircraft moment is the summation of all moments, therefore:

$$M_e = M_w + M_{fus} + M_{ht} + M_{vt} + M_{nac} + M_{eng} + M_{uc} + M_{sur} + M_{sys} + M_{furn} \tag{3.76}$$

For the forward CG, the following cases are selected. Each case without/with fuel:

1. Empty aircraft position.
2. Full payload.
3. Simplified window-seating rule (forward seats only).

Empty aircraft CG position without fuel is:

$$X_{cg_e} = \frac{M_e}{m_e} \tag{3.77}$$

11. Fuel moment:-

$$M_{fuel} = \left(l_{f_{qc}} + x_{l_{eng}} \times tan\Lambda\right) \times m_{fuel} \tag{3.78}$$

Empty aircraft CG position with fuel is:

$$X_{cg_{fe}} = \frac{M_e + M_{fuel}}{m_e + m_{fuel}} \quad (3.79)$$

For the second case,

12. Full Payload moment:-

$$M_{pay} = \left(l_{1_{fus}} + 0.5 \times l_{2_{fus}}\right) \times m_{pay} \quad (3.80)$$

Aircraft CG position without fuel is:

$$X_{cg_{zff}} = \frac{(M_e + M_{pay})}{(m_e + m_{pay})} \quad (3.81)$$

Aircraft CG position with fuel is:

$$X_{cg_{fff}} = \frac{(M_e + M_{pay} + M_{fuel})}{(m_e + m_{pay} + m_{fuel})} \quad (3.82)$$

For the third case, window seats payload mass is:-

$$m_{wpay} = \frac{m_{pay}}{N_{row}} \quad (3.83)$$

13. Forward window seats payload moment:-

$$M_{fw} = \left(l_{1_{fus}} + 0.25 \times l_{2_{fus}}\right) \times m_{wpay} \quad (3.84)$$

Aircraft CG position without fuel is:

$$X_{cg_{zsf}} = \frac{(M_e + M_{fw})}{(m_e + m_{wpay})} \quad (3.85)$$

Aircraft CG position with fuel is:

$$X_{cg_{fsf}} = \frac{(M_e + M_{fw} + M_{fuel})}{(m_e + m_{wpay} + m_{fuel})} \quad (3.86)$$

Forward CG is determined as the largest CG value of all cases above.

For the aft CG position, two cases are selected. Each case without/with fuel:

1. Aircraft at maximum take-off weight with 20% payload at rear of cabin (rear luggage hold).
2. Simplified window-seating rule (rear seats only).

For the first case without fuel, the aft CG position is:

$$X_{cg_{zfa}} = \frac{\left(l_{1_{fus}}+0.5\times l_{2_{fus}}\right)\times 0.8\times m_{pay}+\left(l_{1_{fus}}+l_{2_{fus}}\right)\times 0.2\times m_{pay}+M_e}{m_e+m_{pay}} \quad (3.87)$$

With fuel, the aft CG position is:

$$X_{cg_{ffa}} = \frac{\left(l_{1_{fus}}+0.5\times l_{2_{fus}}\right)\times 0.8\times m_{pay}+\left(l_{1_{fus}}+l_{2_{fus}}\right)\times 0.2\times m_{pay}+M_e+M_{fuel}}{m_e+m_{pay}+m_{fuel}} \quad (3.88)$$

For the second case,

14. Aft window seats payload moment:-

$$M_{aw} = \left(l_{1_{fus}} + 0.75 \times l_{2_{fus}}\right) \times m_{wpay} \quad (3.89)$$

Aft CG position without fuel is:

$$X_{cg_{zsa}} = \frac{(M_e+M_{aw})}{(m_e+m_{wpay})} \quad (3.90)$$

Aft CG position with fuel is:

$$X_{cg_{fsa}} = \frac{(M_e+M_{aw}+M_{fuel})}{(m_e+m_{wpay}+m_{fuel})} \quad (3.91)$$

Aircraft aft CG is determined as the lowest CG value of all cases above.

3.3.4 Aerodynamics

The determination of static stability requires aircraft lift coefficient and aircraft aerodynamic centre to be known. Five major parameters are required to be computed: zero-lift drag, aircraft lift, lift-induced drag, total drag, and stall speed.

3.3.4.1 Zero-lift drag

This parameter is determined as the sum of effects from each aircraft component, plus the effect of the interference at the wing/fuselage junction as follows. Flat plate analogy with correction for section thickness is applied to calculate the wing, horizontal tail, and vertical tail effects. The following coefficients are referenced to the gross wing area:-

1. Wing:

$$C_{d0_w} = 2 \times C_{f_w} \times \left(1 + 2 \times t_{c_w} + 120 \times {t_{c_w}}^4\right) \times \frac{S_{x_w}}{S_w} \tag{3.92}$$

2. Horizontal tail:

$$C_{d0_{ht}} = 2 \times C_{f_{ht}} \times \left(1 + 2 \times t_{c_{ht}} + 120 \times {t_{c_{ht}}}^4\right) \times \frac{S_{ht}}{S_w} \tag{3.93}$$

3. Vertical tail:

$$C_{d0_{vt}} = 2 \times C_{f_{vt}} \times \left(1 + 2 \times t_{c_{vt}} + 120 \times {t_{c_{vt}}}^4\right) \times \frac{S_{vt}}{S_w} \tag{3.94}$$

For fuselage and nacelle, the usual axis symmetric body pressure drag term is used:-

4. Fuselage:

$$C_{d0_{fus}} = C_{f_{fus}} \times \left(1 + \frac{60}{\left(l_{fus}/D_{fus}\right)^3} + 0.0025 \times \frac{l_{fus}}{D_{fus}}\right) \times \frac{S_{w_{fus}}}{S_w} \tag{3.95}$$

5. Nacelle:

$$C_{d0_{nac}} = C_{f_{nac}} \times \left(1 + \frac{60}{(l_{nac}/D_{nac})^3} + 0.0025 \times \frac{l_{nac}}{D_{nac}}\right) \times \frac{S_{w_{nac}}}{S_w} \tag{3.96}$$

Skin friction coefficients in the above equations are determined by applying the Prandtl-Schlichting theory for a fully developed boundary-layer using a roughness factor to account for deviations from the theoretical shape and other surface irregularities as follows:-

$$C_{f_w} = r_w \times 0.455/\left(log_{10}(N_{rey})\right)^{2.58} \tag{3.97}$$

$$C_{f_{ht}} = r_{ht} \times 0.455/\left(log_{10}\left(N_{rey} \times (\bar{c}_{ht}/\bar{c})\right)\right)^{2.58} \tag{3.98}$$

$$C_{f_{vt}} = r_{vt} \times 0.455/\left(log_{10}\left(N_{rey} \times (\bar{c}_{vt}/\bar{c})\right)\right)^{2.58} \tag{3.99}$$

$$C_{f_{fus}} = r_{fus} \times 0.455/\left(log_{10}\left(N_{rey} \times \left(l_{fus}/\bar{c}\right)\right)\right)^{2.58} \tag{3.100}$$

$$C_{f_{nac}} = r_{nac} \times 0.455/\left(log_{10}\left(N_{rey} \times (l_{nac}/\bar{c})\right)\right)^{2.58} \tag{3.101}$$

The wing-fuselage interference zero-lift drag coefficient is determined as follows:-

$$C_{d0_{int}} = 6.75 \times C_{f_{int}} \times w_{TCTR} \times {c_{r_w}}^2/S_w \tag{3.102}$$

Where:-

$$C_{f_{int}} = 0.455/\left(log_{10}\left(N_{rey} \times \left(l_{p_w}/\bar{c}\right)\right)\right)^{2.58} \tag{3.103}$$

Summing zero-lift drag coefficients to determine aircraft zero-lift drag coefficient:

$$C_{d0_{AC}} = C_{d0_w} + C_{d0_{ht}} + C_{d0_{vt}} + C_{d0_{fus}} + C_{d0_{nac}} + C_{d0_{int}} \tag{3.104}$$

3.3.4.2 Aircraft lift

The common way to determine aircraft lift is the ‘aircraft-less-tail’ (AC-T) and adding the tail effect, i.e.:

$$C_{L_{AC}} = C_{L_{AC-T}} + C_{L_{ht}} \times \frac{S_{ht}}{S_w} \tag{3.105}$$

Determination of AC-T lift coefficient is done by summing the effects from each component and modifying the total to account for mutual interference effects. Due to the fact that the aircraft can be flown at different angles of attack (AOA) in various flight conditions, it is easier to determine the lift curve slopes for aircraft components and multiply by the appropriate AOA at each operational point as in the following:

$$\left(\frac{dC_L}{d\alpha}\right)_{AC-T} = K_{int} \times \left(\frac{dC_L}{d\alpha}\right)_{wx} \times \frac{S_{x_W}}{S_W} + \left(\frac{dC_L}{d\alpha}\right)_{fus} \times \frac{S_{cs_{fus}}}{S_W} + N_{eng} \times \left(\frac{dC_L}{d\alpha}\right)_{nac} \times \frac{S_{cs_{nac}}}{S_W} \tag{3.106}$$

Where, K_{int} is the wing-fuselage interference effect factor. For a conventional circular fuselage:

$$K_{int} = 1 + 2.25 \times \frac{D_{fus}}{b_w} \tag{3.107}$$

The Exposed Wing lift curve slope is computed as:-

$$\left(\frac{dC_L}{d\alpha}\right)_{wx} = f \times \left(\frac{C_{L_W}}{E + \frac{C_{L_W}}{A_W \times \pi}}\right) \tag{3.108}$$

Where, $f = 0.995$ for conventional tapered wings, and E is the Jones velocity factor. For straight tapered wings defined as:

$$E = 1 + \frac{2 \times \lambda_W}{A_W \times (1 + \lambda_W)} \tag{3.109}$$

Fuselage and Nacelle lift curve slopes are determined as:

$$\left(\frac{dC_L}{d\alpha}\right)_{fus} = 2 \times (K_2 - K_1)_{fus} \tag{3.110}$$

$$\left(\frac{dC_L}{d\alpha}\right)_{nac} = 2 \times (K_2 - K_1)_{nac} \tag{3.111}$$

Where: $(K_2 - K_1)$ is computed using **Figure 3-1** [90], which represents the graph of $(K_2 - K_1)$ against body length/diameter ratio.

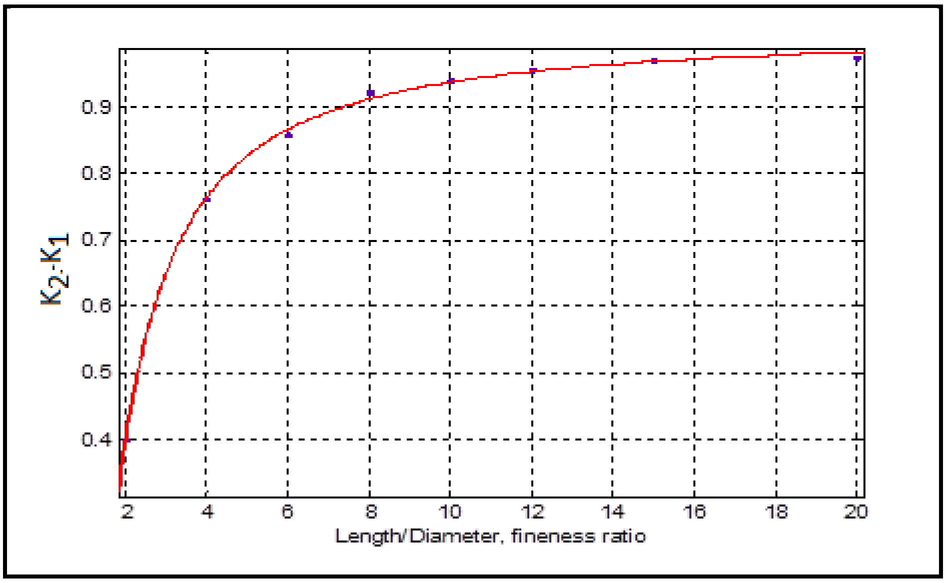

Figure 3-1: (K_2-K_1) versus body fineness ratio, [90]

Now, referring to Equation (3.105), the horizontal tail lift ($C_{L_{ht}}$) is computed as:

$$C_{L_{ht}} = \frac{C_{m_{AC}}}{V_{ht}} + \frac{C_{L_{AC}}}{V_{ht}} \times \left(\frac{X_{CG}-X_{AC}}{\bar{c}}\right) \tag{3.112}$$

To compute Equation (3.112), the aerodynamic centre and the volume coefficients are required, which can be computed as follows:

The distance of the aircraft aerodynamic centre from the nose reference point(X_{AC}) is determined as:

$$X_{AC} = X_{ac_w} + X_{ac_{fus}} + X_{ac_{nac}} \tag{3.113}$$

Where: $X_{ac_w} = l_{f_{ac}}$ as calculated in the geometry section.

$$X_{ac_{fus}} = -\left(\frac{1.8 \times D_{fus}^2 \times l_{p_W}}{\left(\frac{dC_L}{d\alpha}\right)_{AC-T} \times S_W}\right) \tag{3.114}$$

$$X_{ac_{nac}} = K_{nac} \times \left(\frac{D_{nac}^2 \times l_{nac_{le_{\bar{c}/4}}}}{\left(\frac{dC_L}{d\alpha}\right)_{AC-T} \times S_W}\right) \times N_{eng} \tag{3.115}$$

Where: $K_{nac} = -4$ for wing mounted

$K_{nac} = 2.5$ for rear fuselage mounted

The aircraft pitching moment coefficient ($C_{m_{AC}}$) is determined as:

$$C_{m_{AC}} = C_{m_w} + C_{m_{fus}} \tag{3.116}$$

Where: C_{m_w} is a design input variable

Fuselage pitching moment coefficient is computed using Munk's theory as:

$$C_{m_{fus}} = -\left(\frac{1.8\times(1-2.5\times D_{fus}/l_{fus})\times\pi\times D_{fus}^2\times l_{fus}}{S_w\times\bar{c}\times 4}\right)\times w_{inc} \tag{3.117}$$

Hence:

$$C_{L_{AC-T}} = C_{L_{AC}} - C_{L_{ht}} \times \frac{S_{ht}}{S_w} \tag{3.118}$$

The angle of attack of an aircraft is determined as:

$$\alpha = \left(C_{L_{AC-T}} + \left(\frac{dC_L}{d\alpha}\right)_{fus} \times w_{inc}\right) \Big/ \left(\frac{dC_L}{d\alpha}\right)_{AC-T} \tag{3.119}$$

Then, lift from each aircraft component is:

$$C_{L_{nac}} = \left(\frac{dC_L}{d\alpha}\right)_{Nac} \times \alpha \tag{3.120}$$

$$C_{L_{fus}} = \left(\frac{dC_L}{d\alpha}\right)_{Fus} \times \alpha \tag{3.121}$$

$$C_{L_w} = C_{L_{AC-T}} - C_{L_{nac}} - C_{L_{fus}} \tag{3.122}$$

3.3.4.3 Lift induced drag

Aircraft lift induced drag is determined as the sum of the aircraft component effects plus the interference effect. i.e.:

$$C_{di_{AC}} = C_{di_w} + C_{di_{ht}} + C_{di_{fus}} + C_{di_{nac}} + C_{di_{int}} + w_{\Delta cd} \tag{3.123}$$

Where:

$C_{di_w} = C_{L_w}^2 \times \frac{1-\delta}{\pi \times A_w}$, where: δ is a parameter evaluated from a graph of A_w and λ given in [91] and curve fitted in the program. (3.124)

$C_{di_{ht}} = C_{L_{ht}}^2 \times \frac{f_o}{\pi \times A_{ht}} \times \frac{S_{ht}}{S_w}$, where: f_o is Oswald factor ≈ 1.2 for horizontal tail. (3.125)

$$C_{di_{fus}} = C_{L_{fus}} \times (\alpha - w_{inc}) \quad (3.126)$$

$$C_{di_{nac}} = C_{L_{nac}} \times \alpha \quad (3.127)$$

Interference coefficient ($C_{di_{int}}$) is determined using Munk's Stagger theorem as:

$$C_{di_{int}} = \frac{2 \times \sigma}{\pi} \times C_{L_w} \times C_{L_w} \times \frac{S_{ht}}{b_w \times b_{ht}} \quad (3.128)$$

Where: σ is determined from curve fitted data with downwash gradient and ratio ($\frac{b_{ht}}{b_w}$) as the coordinates [92].

The last component in aircraft lift induced drag ($w_{\Delta cd}$) is the increment in profile drag coefficient due to angle of attack. It is calculated as:

$$w_{\Delta cd} = \frac{0.75 \times x}{(C_{Lmax} - C_{Ld})^2} \times \left(C_{L_W} - C_{Ld}\right)^2 \quad (3.129)$$

Where: $x = 67 \times \frac{C_{Lmax}}{\left(log_{10}(N_{rey})\right)^2} - 0.0046 \times \left(1 + 2.75 \times t_{c_w}\right)$ (3.130)

Hence, the total aircraft drag is:

$$C_{d_{AC}} = C_{d0_{AC}} + C_{di_{AC}} \quad (3.131)$$

3.3.4.4 Stall speed

Stall speed (at take-off and at landing) is an important parameter since it indicates the slowest speed at which an aircraft can travel and generate enough lift to remain or become airborne. It depends mainly on aircraft configuration, state of flaps, and other lift-control devices. Take-off stall speed is computed as:

$$V_{st} = \sqrt{\frac{m_{to} \times 9.81}{0.5 \times \rho \times S_W \times C_{L_{max}}}} \tag{3.132}$$

While landing stall speed is computed as:

$$V_{sl} = \sqrt{\frac{m_{lan} \times 9.81}{0.5 \times \rho \times S_W \times C_{L_{max}}}} \tag{3.133}$$

3.3.5 Stability

Simple stability calculations are necessary to determine the static margin. The static margin is evaluated as:

$$static\ margin = \frac{X_{NP} - X_{CG}}{\bar{c}} \tag{3.134}$$

Where X_{CG} is the aft CG position and X_{NP} is the position of the aircraft neutral point which defined as:

$$X_{NP} = X_{AC} + \bar{c} \times \left(\frac{\left(\frac{dC_L}{d\alpha}\right)_{ht}}{\left(\frac{dC_L}{d\alpha}\right)_{AC}} \right) \times \left(1 - \frac{d\varepsilon}{d\alpha}\right) \times V_{ht} \tag{3.135}$$

Where X_{AC} as calculated in equation (3.113), and

$\left(\frac{dC_L}{d\alpha}\right)_{ht}$ is the lift curve slope of the horizontal tail and determined as:

$$\left(\frac{dC_L}{d\alpha}\right)_{ht} = f \times \left(\frac{C_{L_W}}{E + \frac{C_{L_W}}{A_{ht} \times \pi}}\right) \tag{3.136}$$

Where $f = 0.995$ for conventional tapered wings, and assume same 2-D slope as wing. E is the Jones velocity factor. For straight tapered wings defined as:

$$E = 1 + \frac{2 \times \lambda_{ht}}{A_{ht} \times (1 + \lambda_{ht})}$$

$\left(\frac{dC_L}{d\alpha}\right)_{AC}$ is the lift curve slope of the whole aircraft which is defined as:

$$\left(\frac{dC_L}{d\alpha}\right)_{AC} = \left(\frac{dC_L}{d\alpha}\right)_{AC-T} + \left(\frac{dC_L}{d\alpha}\right)_{ht} \times \frac{S_{ht}}{S_w} \times \left(1 - \frac{d\varepsilon}{d\alpha}\right) \tag{3.137}$$

$\frac{d\varepsilon}{d\alpha}$ is the downwash gradient which is defined as:

$$\frac{d\varepsilon}{d\alpha} = \frac{1.75 \times \left(\frac{dC_L}{d\alpha}\right)_W}{\pi \times A_W \times \left(\lambda_W \times 2 \times \frac{l_{ht}}{b_W}\right)^{0.25} \times (1 + f_m)} \tag{3.138}$$

Where f_m : is the factor which depends on the flow separation from both tail and wing surfaces. It is computed for high and low positions of wing and tail as follows:

$$f_m = 2 \times \left(\frac{D_{fus}}{2} + l_{ht} \times sin(w_{inc})\right) / b_w \quad \text{, for low wing, low tail} \tag{3.139}$$

$$f_m = 2 \times \left(\frac{3 \times D_{fus}}{4} + l_{ht} \times sin(w_{inc}) + b_{vt}\right) / b_w \quad \text{, for low wing, high tail} \tag{3.140}$$

$$f_m = 2 \times \left|\left(\frac{-D_{fus}}{2} + l_{ht} \times sin(w_{inc})\right)\right| / b_w \quad \text{, for high wing, low tail} \tag{3.141}$$

$$f_m = 2 \times \left(b_{vt} - \frac{D_{fus}}{4} + l_{ht} \times sin(w_{inc})\right) / b_w \quad \text{, for high wing, high tail} \tag{3.142}$$

3.3.6 Performance

Aircraft performance analysis in the preliminary design stage consists of flight profile analysis and field performance.

3.3.6.1 Flight profile performance

Climb rate and descent rate are two important parameters that are considered in the field profile performance. Best climb performance is achieved when the total available thrust is more than aircraft drag. Civil aircraft make an en route climb to cruise altitude at a quasi-steady-state climb by holding the climb speed at constant Mach number. Since the parameters of climb performance vary with altitude, it is typically computed in discreet steps of altitude, i.e. all parameters are considered invariant and taken as an average value within the altitude steps. "*The engineering approach is* to *compute the integrated distance covered, the time taken, and the fuel consumed to reach the cruise altitude in small increments and then totalled.*" [93]. Torenbeek's equation to compute the rate of climb is:

$$R_c = \frac{V_{AC} \times \left(T_{AC} - C_{d_{AC}}\right)}{m_{AC} \times \left(1 + 0.567 \times N_{ma}^2\right)} \tag{3.143}$$

The rate of descent is the opposite of the rate of climb. Therefore, it has a negative sign since the drag is higher than thrust. It uses the same approach and equations as for a climb.

3.3.6.2 Field performance

Three major parameters are considered in this category which are: Balanced Field Length (BFL), Landing Field Length (LFL), and Second Segment Climb Gradient. Takeoff is achieved when an aircraft accelerates under maximum thrust until a suitably safe speed is attained: at this point lift equals the weight of the aircraft. For safety purposes, mandatory requirements for taking-off with one engine inoperative (OEI) are to clear an obstacle of 35-ft height. BFL is computed when the stopping distance after an engine failure at decision speed is the same distance taken to clear the obstacle at maximum takeoff weight. Torenbeek's equation for determination of BFL is:

$$B_{FL} = \left(\frac{0.704}{1+2.3\times(G_{seg2}-0.024)}\right)\times\left(\frac{m_{to}}{S_w\times C_{L_{v2}}}+14\right)\times\left(\frac{9.81\times m_{to}}{0.8\times T_{eng}\times N_{eng}-0.04}+2.7\right)+200 \tag{3.144}$$

Where: minimum airworthiness gradient is 0.024

Obstacle height is the metric equivalent of 50 ft, i.e. 14 m

runway friction coefficient is 0.04

typical inertia distance is 200 m

Landing performance is similar to takeoff performance with full flaps extended and the aircraft at landing weight. The difference is that a landing phase after touchdown requires the aircraft to decelerate rapidly. Thrust reversers provide negative thrust as a decelerating force. Aircraft drag is higher than at takeoff due to flap extension and air brakes. LFL is determined based on Loftin's analysis [94] as:

$$L_{FL} = \left(5.612\times\frac{m_{lan}}{S_w\times C_{Lm_{tl}}}+153\right)\times 1.667 \tag{3.145}$$

Where: 1.667 is the required airworthiness multiplying factor

The climb rate after the screen height of 35ft, is known as the second segment climb gradient (G_{seg2}), which needs to be evaluated for the OEI case, with one engine inoperative. This parameter is determined at ISA conditions and/or WAT limit conditions. For WAT calculations, ambient temperature and airfield height should be available as input design variables. The basic drag of the aircraft is determined in the Aerodynamics section (3.3.4). Due to asymmetric flight following an engine failure, the drag is increased. It can be estimated as described by Torenbeek [10]:

$$\Delta C_d = \Delta C_{d_e} + \Delta C_{d_{as}} \tag{3.146}$$

Where ΔC_{d_e} is the drag increment due to wind milling effect and is determined as:

$$\Delta C_{d_e} = 0.0785\times 0.7225\times D_{nac}^2 + \frac{0.1472\times0.7854\times0.36\times D_{nac}^2}{1+0.16\times N_{ma}^2} \tag{3.147}$$

The increment in drag of the aircraft due to asymmetric flight is $\Delta C_{d_{as}}$ and can be estimated by the following Torenbeek's equation,

$$\Delta C_{d_{as}} = K_s \times \left(\frac{T_{eng} \times (N_{eng} - 1)}{0.5 \times \rho \times v^2} + C_{d_e}\right)^2 \tag{3.148}$$

Where,

$$K_s = \left(\frac{x_{l_{eng}}}{l_{vt}}\right)^2 \times \frac{1}{S_{vt} \times \pi \times A_{vt}} \times \left(1 + 2.3 \times 0.67 \times \left(\frac{A_{vt}}{cos\Lambda_v}\right)^{-0.33}\right) \tag{3.149}$$

Another increase in induced drag is due to the uneven distribution of wing span wise loading which is determined as:

$$\Delta C_{d_i} = \frac{0.1 \times f_o \times C_{L_{to}}^2}{\pi \times A_w} \tag{3.150}$$

Where, f_o is the Oswald factor ≈ 1.5

Hence, the total drag is:

$$C_{d_{tot}} = 0.5 \times \rho \times v^2 \times S_w \times \left(C_{d_{v2}} + \Delta C_{d_i} + \frac{C_{d_e} + C_{d_{as}}}{S_w}\right) \tag{3.151}$$

The Second climb gradient may now be computed as:

$$G_{seg2} = \frac{T_{eng} \times (N_{eng} - 1) - C_{d_{tot}}}{m_{to} \times 9.81} \tag{3.152}$$

3.3.7 Cost

Cost-benefit analysis is a significant parameter in evaluating competitive aircraft designs as it is based on the minimum cost per unit work performed. Aircraft cost consists of airframe cost and engine cost, i.e.:

$$C_{AC} = C_{AF} + C_{eng} \times N_{eng} \tag{3.153}$$

The rule of thumb is airframe cost depends mainly on aircraft empty weight, i.e.:

$$C_{AF} = m_e \times C_{AFc} \tag{3.154}$$

While engine price is based on its takeoff thrust as

$$C_{eng} = T_{eng} \times C_{engc} \tag{3.155}$$

Aircraft price is not the only parameter under consideration, but also aircraft operating costs. These operating costs are classified into two categories which are direct operating cost (DOC) and indirect operating cost (IOC). IOC is difficult to estimate well, since it depends on the services that the airline (customer) offers. Therefore, DOC is a useful and widely-used parameter for comparative analysis.

Several methodologies [95][96][97] have been used in the past to estimate DOC. Kundu [93] suggests using AEA [98] methodology, which is widely used in Europe, as a standard approach to identify DOC. The main components of DOC are:

1- Depreciation.
2- Insurance.
3- Interest on capital.
4- Crew salary and staff wages.
5- Airframe maintenance, labour and material.
6- Engine maintenance, labour and material.
7- Landing fees.
8- Navigational charges.
9- Ground-handling charges.
10- Fuel charges.

The following equations are used to estimate the value of each component. Note that these equations compute the component DOC per block hour. To determine a flight cost, the DOC per block hour is multiplied by the block time:

1- Depreciation: This is calculated over a 14 year period to a residual value of 10%. It is determined as:

$$C_{dp} = \frac{0.9 \times (C_{AC} + 0.1 \times C_{AF} + 0.3 \times C_{eng} \times N_{eng}) \times t_B}{14 \times U} \tag{3.156}$$

Where: U is the utilisation (calculated per block hour per annum in hours/year):

$$U = \left(\frac{3750}{t_B + 0.5}\right) \times t_B \tag{3.157}$$

2- Insurance cost:

$$C_{ins} = \frac{0.005 \times C_{AC}}{U} \tag{3.158}$$

3- Interest cost:

$$C_{int} = \frac{0.053 \times (C_{AC} + 0.1 \times C_{AF} + 0.3 \times C_{eng} \times N_{eng}) \times t_B}{U} \tag{3.159}$$

4- Flight crew costs \$493 per block hour for a two-crew operation, while \$81 per block hour for each cabin crew member.

5- Airframe maintenance cost:

$$C_{am} = C_{al} + C_{amm} \tag{3.160}$$

Where:

$$C_{al} = \left(0.09 \times \frac{m_{e-e}}{1000} + 6.7 - \frac{350}{\left(\frac{m_{e-e}}{1000}+75\right)}\right) \times \left(\frac{0.8+0.68\times(t_B-0.25)}{t_B}\right) \times C_{lr} \tag{3.161}$$

and,

$$C_{amm} = \left(\frac{4.2+2.2\times(t_B-0.25)}{t_B}\right) \times C_{AC} \times 10^{-6} \tag{3.162}$$

6- Engine maintenance cost:

$$C_{em} = N_{eng} \times (C_{el} + C_{emm}) \times \frac{t_B+1.3}{t_B-0.25} \tag{3.163}$$

Where:

$$C_{el} = 0.21 \times C_{lr} \times \left(1.27 - 0.2 \times {E_{bpr}}^{0.2}\right) \times (0.032 \times N_c + k) \times \left(1 + \frac{T_{eng}}{1000}\right)^{0.4} \tag{3.164}$$

and

$$C_{emm} = 2.56 \times \left(1 + \frac{T_{eng}}{1000}\right)^{0.8} \times \left(1.27 - 0.2 \times {E_{bpr}}^{0.2}\right) \times \left(0.4 \times \left(E_{oapr}/20\right)^{1.3} + 0.4 + 0.032 \times N_c + k\right) \tag{3.165}$$

Where: k = 0.5 for a single spool engine

k = 0.57 for a double spool engine

k = 0.64 for a triple spool engine

7- Landing fees:

$$C_{lf} = \frac{7.8}{t_B} \times \frac{m_{to}}{1000} \tag{3.166}$$

8- Navigational charges:

$$C_{nav} = \left(\frac{0.5\times s_l}{1000}\right) \times \sqrt{\frac{m_{to}}{50000}} \tag{3.167}$$

9- Ground-handling charges:

$$C_{grd} = \frac{100 \times m_{pay}}{t_B \times 1000} \tag{3.168}$$

10- Fuel charges:

$$C_{fuel} = \frac{m_{fuel} \times C_{fu}}{\rho_f} \tag{3.169}$$

Where: m_{fuel} is in pounds (lbs) and excluding reserves, while $\rho_f = 6.7\ lbs/gal.$

The DOC can then be calculated by summation of Equations (3.156) to (3.169).

3.4 Conclusions

Design of an aircraft requires many variables to be computed, mainly to arrive at a design that fulfils most of the mission objectives. The effect of changes in one or more variables may not be obvious to the beginner; therefore an iterative process is usually followed. Whilst some variables are selected, others are calculated. As the complexity of the design is increased, the number of variables that reflect the design choices is also increased. In the preliminary design phase, the input and output variables are grouped according to aircraft components and mission requirements. All equations required for analysing the output variables, in this phase were also grouped into sections for clarity and easy understanding. Torenbeek's formulae are used in the development process of the iADS software due to their accuracy. The next chapter shows how the design methodology is coded into an interactive design software tool.

Chapter FOUR

Aircraft Design Software Implementation

4.1 Introduction

Implementing new aircraft design software for teaching is not simple. It needs proper consideration and effort to overcome all the limitations and disadvantages of the existing software plus additional features that are necessary for teaching. Due to student time limitations, the software should facilitate student understanding of different aspects of aircraft design and to speed up the design process as well. The software should have a friendly-user interface, comprehensive analysis without making it overtly complex, parametric studies for sensitivity analysis and an optimiser to fine tune the design, etc. as mentioned at the end of chapter two. This chapter describes in some detail the implemented software, which has been developed using a modular approach. Each aspect of design is realised through an appropriately named module. A module in this context is an object that takes information from other linked objects and passes the results to any other object that needs the information.

4.2 Graphical user interface (GUI)

A GUI is the way a user interacts with the computer and it is almost as important as the software computations itself. As the computing and graphical processing ability of computers has grown, with it has grown the graphical user interfaces. Actually, it dresses up the software functions and allows access to software functionality in a cohesive manner. If the GUI is too complex and/or contains many symbolic (ambiguous) items, it confuses the user and the software results may be erroneous due to inappropriate interactions. GUI design should be based on the knowledge of the user's experiences and expectations. In the present context, since the user of the Interactive Aircraft Design Software (iADS) is going to be an undergraduate student, it should be in keeping with his/her cognitive abilities.

However, a GUI is based on the integration of a number of elements that bring the tasks and work we do on the computer to life. In other words, a GUI is the graphical representation of, and interaction with, programs, data, and objects on the computer screen. A GUI is WIMP, which means that GUI is made of these elements: Windows, Icons, Menus, and Pointers. Another way to define a GUI is WYSIWYG, what you see is what you get. The main features of a GUI outlined by Galitz [99] are as follows:

1. *Symbols are recognised faster than text.*
2. *Faster learning.*
3. *Faster use and problem solving.*
4. *Easier remembering.*
5. *More natural.*
6. *Exploits visual/spatial cues.*
7. *Foster more concrete thinking.*
8. *Provide context.*
9. *Fewer errors.*
10. *Increase feeling of control.*
11. *Immediate feedback.*
12. *Predictable system responses.*
13. *Easily reversible actions.*
14. *Less anxiety concerning use.*
15. *More attractive.*
16. *May consume less space.*
17. *Replaces national languages.*
18. *Easily augmented with text displays.*
19. *Low typing requirements.*
20. *Smooth transition from command language system.*

One of the limitations of a GUI environment developed on the Windows platform is its application specific interface. Various sets of information that the user wishes to manipulate are delivered through an application specific interface. Initially, users must start an application before doing anything with data that is stored in files. The action of double-clicking the application icon starts the application. The interface to the functionality of the application is presented via a set of menus that are grouped by functionality. *Most of what users see in the Windows environment are mainly application-oriented interfaces (or function-oriented interface),* [100].

In 1992, IBM released a new generation of a GUI based on object-oriented architecture. "*Object-oriented interfaces are sometimes described as turning the application inside-out as compared to function-oriented interfaces. The main focus of the interaction changes to become the users' data and information objects that are typically represented graphically on the screen as icons or in windows*" [101]. The difference between a GUI and an object-oriented user interface (OOUI) is very difficult to explain without actually using them since OOUIs include all the features of GUIs. A key characteristic of an OOUI is that it fights to overcome the main drawback of GUI, i.e. its application orientation. The following example explains this difference:

Figure 4-1 [100], shows the new clock in Windows 95 Power Toys developed by Microsoft. It is based on a GUI. The user can change the aspects of the clock by using the properties in the drop-down menu. **Figure 4-2** [100], shows the OS/2 system clock. It is an OOUI. The application window opens into the default view, Date/Time. The user can open another view of the clock where the properties available in a multipage notebook, i.e., multiple objects of the same type can co exist which, are differentiated by its properties. For example, one instance of the clock object could show an analogue representation and the other digital. Depending upon the set of properties, a change in one object instance can automatically be reflected in the other object.

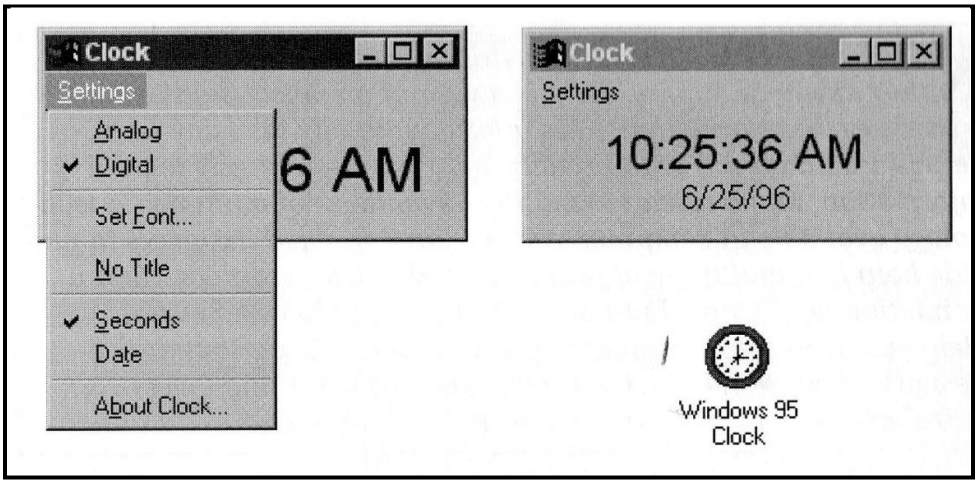

Figure 4-1: GUI application, [100]

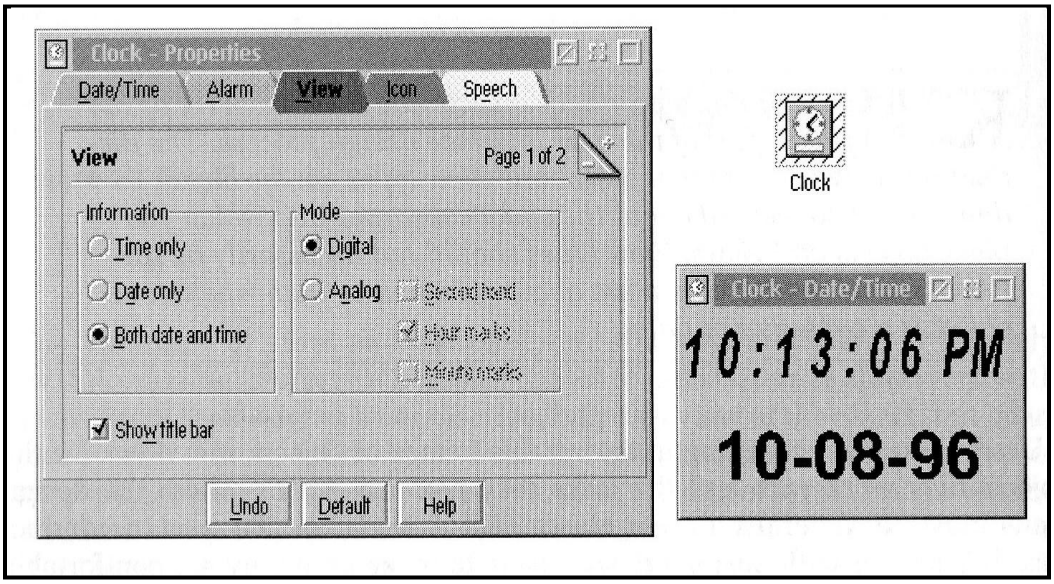

Figure 4-2: OOUI application, [100]

The benefits of using GUIs are many, if the software packages are well-designed. Without a well-designed GUI, even a software package with outstanding features will not be successful or be used effectively. GUIs can educate and entertain as well as allow users to do their work. All GUIs of the existing aircraft design software are user friendly-user interfaces that tend to be application-oriented interfaces. Nowadays, as technology (including GUI) has improved, some old style GUIs seem un-friendly and hence lose usability appeal. In order to overcome these drawbacks, an OOUI is designed using object-oriented programming language (Delphi). Though many OOP programming languages exist such as C++ or common LISP to name a few, the choice of Delphi as a development tool was made due to its similarity with FORTRAN, and since many existing codes had to be translated it seemed to be the obvious choice. The iADS software layout is similar to many existing applications, comprising a menu bar window which consists of these elements: File, Edit, Configure, Design, Output, Window, and Help. File, Edit, Window, and Help elements are general purpose menus. The student starts to work with the configuration menu to enter his/her designed variables' values. This functionality is grouped in well-defined items that are easily understood. Each item consists of its related set of design distinct features in aircraft design. These items are: Wing, Fuselage, Tail, Aerodynamics, Propulsion (Engine &

Nacelle), Speeds, Stages, Heights, Weights, and Cost. This configuration is unlikely to confuse the student and facilitates the input process as shown in **Figure 4-3**.

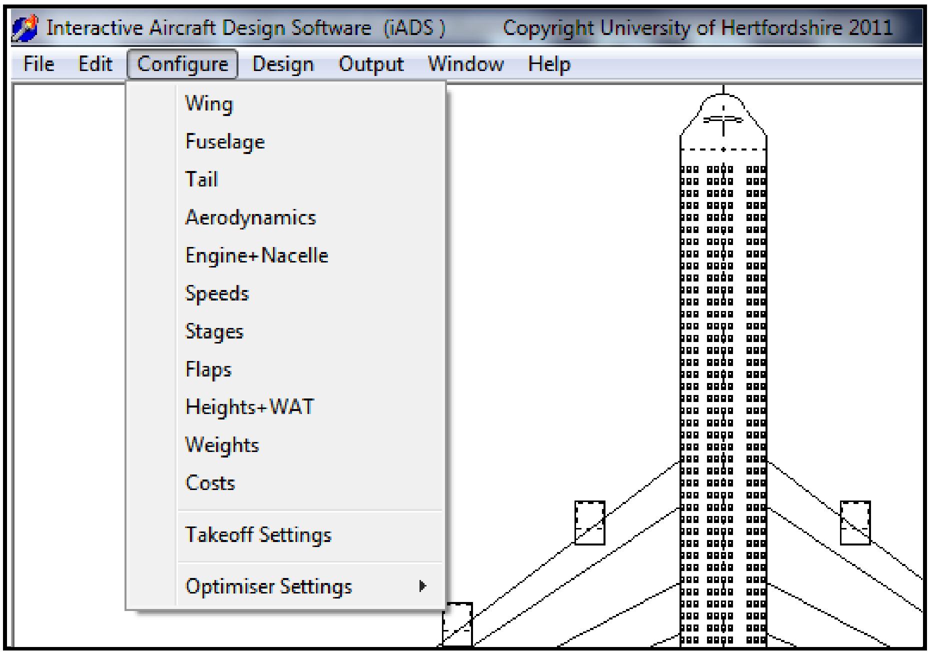

Figure 4-3: Main menu form

Figure 4-4 shows Wing menu item data entry form for Boeing 737-800 aircraft as an example. Investigating the form in general, many features are outlined as follows:

1. A separate window is used (this helps student to focus on a specified area of the screen).
2. The number of variables is kept to a minimum (no more than ten, depending on the item's variables).
3. Two options are available for entering the design variable's value (either direct in the specified space, or through the scroll bar).
4. The presence of the scroll bar limits the input value between lower and upper bounds (this helps the student to understand the variable's range).
5. Design variable names are defined in full with no symbols used (so the student can recognise the variable well).
6. An alert message is popped up by the expert system (if a variable's assigned value is out of range when using direct data entry method).

7. A two dimensional (2D) 3-View of the design aircraft window is popped up that dynamically changes, if the scroll bar method is used to change variables, i.e. an OOUI (to help student to understand the effect of variable variation).
8. Basic colour (black) is used (many colours confuse the student).
9. Status bar is added to show hints for student.

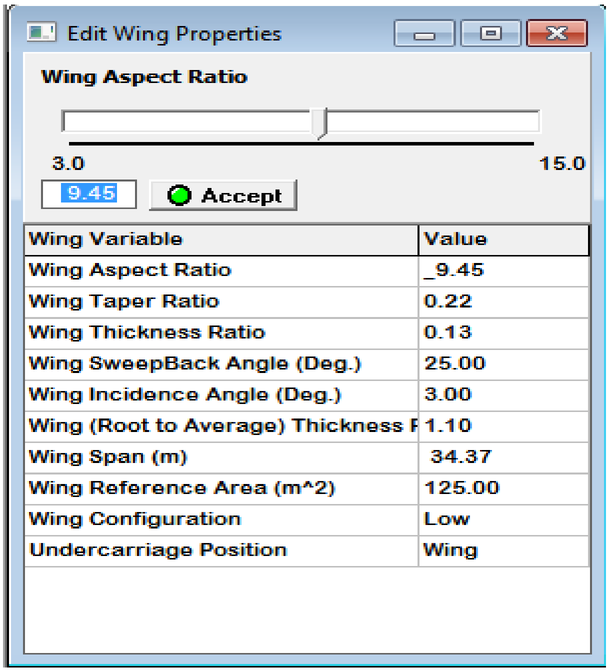

Figure 4-4: Wing menu form

As the student completes the input process, the design stage starts which is invoked by the Design menu, in the menu bar. It consists of three items which are: without optimisation, with optimisation, and dynamic stability. The first item "without optimisation" performs the design iteration once. The output parameters as well as the input design variables are displayed in a tree form window as shown in **Figure 4-5**. The tree has two main roots, i.e. input variables and output parameters. Each root has branches and sub-branches. This type of implementation allows the student to focus on some parameters rather than the whole output results. The value of this feature appears in performing parametric studies on assigned design variables. The second item in the Design menu is "with optimisation" which performs optimisation to fine tune the design. This requires an additional item in the Configure menu to set the value of the free (or fixed) design variables, constraints, and the objective function related the optimisation process as explained later. The last item is “dynamic stability” which opens a new window to perform dynamic stability, also discussed later.

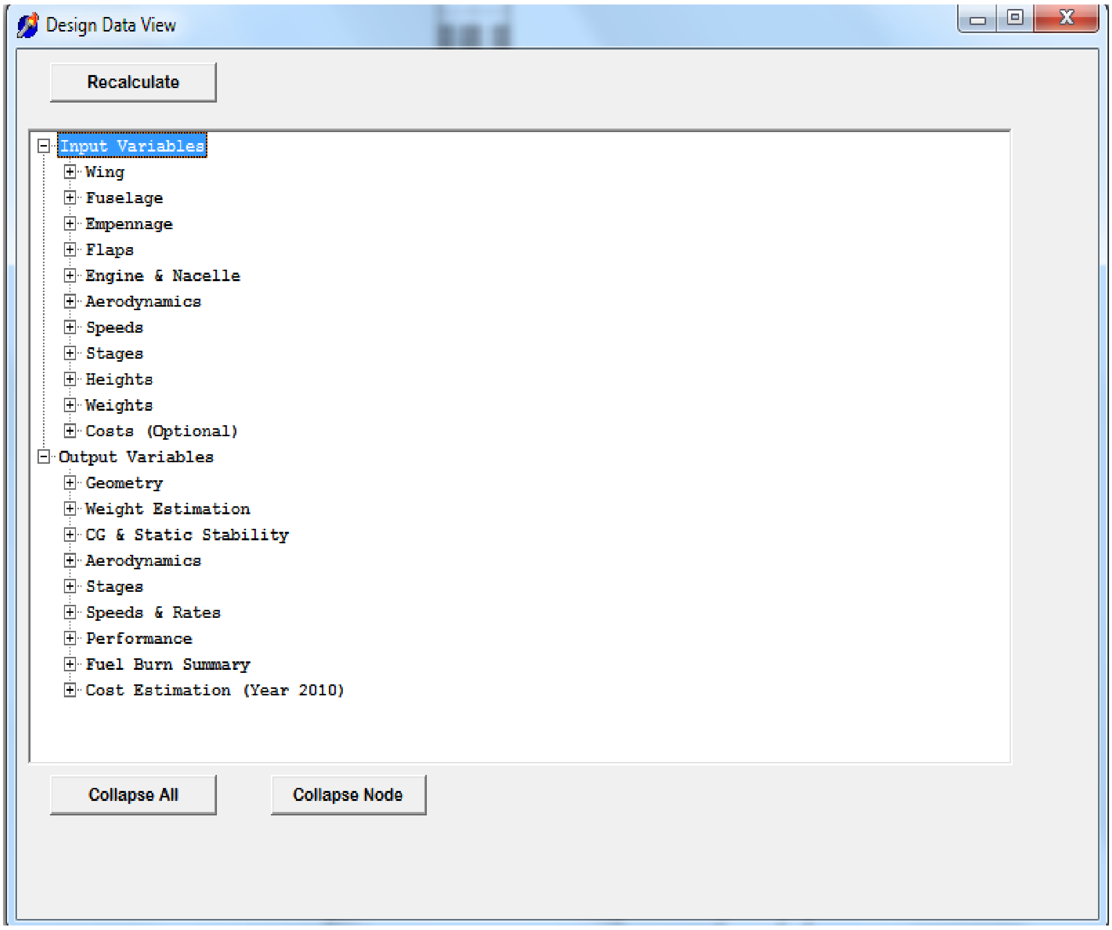

Figure 4-5: “without optimisation” form in the ‘Design’ menu

The “Output” menu in the menu bar also consists of three items which are: text data, 2D 3-view of the aircraft, and parametric studies. The full set of results is listed, organised in an easy to understand format. These results can be saved as a text file or as a spreadsheet for further analysis if required. The “3-view aircraft” item draws the designed aircraft as a 2D 3-view form. This allows the student to visualise the effect of the design variable variation on the shape as mentioned above. The last item is “parametric studies” which performs, as its name refers to, parametric studies between the input design variables and the output parameters in a graphical format to be discussed later.

4.3 Synthesis program

All equations, presented in the previous chapter, are used to develop the synthesis program. Each equation-group is implemented as a module that can be easily edited, modified, or replaced by another module. These modules are: **Geometry, Weight, CG, Aerodynamics, Stability, Flight Performance,** and **Cost Estimation**.

4.3.1 Geometry

The geometry module sizes the fixed aircraft components. This involves the calculation of the major geometries related to the wing, fuselage, empennage, flaps, and nacelles. The wing geometry algorithm starts to calculate wing related parameters such as: Span, Root chord, Tip chord, Mean geometric chord, Mean aerodynamic chord, Root thickness, Exposed span, Exposed area, Exposed aspect ratio, Distance from fuselage nose (Datum point) to wing quarter root, and Distance from datum to wing centre of pressure. For flap related parameters, it calculates Flap-span inboard position, Flap-span outboard position, Flap-inboard-span to wing-span ratio, Flapped wing area, and Flap area. In the fuselage part, three parameters are determined which are: Tail cone length, Fuselage length, and Fuselage wetted area.

The empennage component consists of horizontal and vertical tails. Calculated parameters for both tails are: Area, Root chord, Span, Mean geometric chord, and Tail arm. The tail arm for the vertical tail is calculated based on engine positions (either wing mounted, or rear fuselage mounted), whilst for horizontal tail calculations are based on the tail configuration (either conventional, or T-tail). Nacelle wetted area is the only calculated parameter for the nacelle component. All these parameters are computed based on standard geometrical relationships.

Figure 4-6 shows a sample result for Boeing 737-800 aircraft which also includes input design variables.

```
                         Wing       Tailplane     Fin
Aspect Ratio             9.45        5.54        1.56
Gross Area     (m^2)   125.00       30.53       22.26
Span              (m)   34.37       13.01        5.89
Taper Ratio              0.22        0.18        0.31
Thickness chord ratio 0.13        0.11        0.11
MeanAeroChord     (m)    4.13        2.35        3.78
Chord at C.line  (m)    5.96        3.98        5.77
Tail Arm          (m)   =====       18.64       18.53
Sweep Angle   (deg.)   25.00       =====       31.39

Wing Location 12.93 (m) from nosecone apex to the leading edge

                       Fuselage    One Nacelle
Diameter         (m)     3.89        2.06
Total Length     (m)    38.02        4.70
NoseConeLength (m)       4.00        3.00
Cabin Length     (m)    22.35
TailConeLength (m)      11.67        1.70
Wetted Area   (m^2)    416.76       23.47

Number of Passengers        =  189
Total Take-off Thrust (lbs)=  54376.0
Engin scale factor          =  4.00
```

Figure 4-6: Sample geometry output results

4.3.2 Weights

This is the most significant module due to the fact that accurate weight estimation at early stages is a hard and a difficult process. This module is developed using the common general build-up methodology. It breaks down the maximum takeoff weight into its components and adds them to determine empty, operational empty, zero-fuel, and maximum takeoff weights. Empty weight comprises of the following aircraft components: wing (including flaps), fuselage, empennage, propulsion (engines and nacelles), undercarriage, surface controls, furnishings, and systems (which include APU, electrical, hydraulics, instruments and avionics, and air conditioning and anti-icing). Operating items and crew weights are added to empty weight to form operating empty weight. Adding payload weight to the operating empty weight gives zero-fuel weight. Finally, fuel weight is added to achieve the total maximum takeoff weight. Load factors (limit and ultimate) are calculated at the beginning of this module. Torenbeek's formulae are used and an example result for Boeing 737-800 aircraft is shown in **Figure 4-7**.

```
Load Gust Factor          = 5.092
Manouvre Load Factor      = 4.125
Design Dive Speed (I.A.S.) (m/s) =  292

@@@@@@@@@@@ Mass calculation @@@@@@@@@@@@@@@@@@@@
Wing includes flaps (kg)  =   8834.9
Fuselage            (kg)  =   9151.5
Empennage           (kg)  =   1825.6
Nacelles            (kg)  =   1356.6
Engines             (kg)  =   4743.9
Propulsion System   (kg)  =   6534.1
Propulsion (total)  (kg)  =   7890.7
Undercarriage       (kg)  =   3172.9
Surface Controls    (kg)  =   1224.7

Auxiliary power unit                  (kg) =    174.3
Paint & Oxygen system                 (kg) =    732.3
Electrical system                     (kg) =   1502.5
Avionics & Instruments (+ AutoPilot)  (kg) =   1078.4
Air conditioning & Anti-icing system  (kg) =    746.8
Hydraulic system                      (kg) =    866.3
Systems (Total)     (kg)  =   5100.7

Furnishings         (kg)  =   4550.4
Empty Mass          (kg)  =  41751.3

Operation Items     (kg)  =   1628.6
Crew mass           (kg)  =    186.0
Flight attendants   (kg)  =    408.0
Op. empty mass      (kg)  =  43973.9

Passsenger Load     (kg)  =  22718.4
Zero Fuel mass      (kg)  =  66692.3
Total Fuel          (kg)  =  16500.0

Maximum TakeOff     (kg)  =  83192.3
```

Figure 4-7: Sample output result from the weights module

4.3.3 Centre of gravity (CG)

Good stability and control is a foremost requirement for operational aircraft. To achieve this goal, the evaluation of aircraft CG is required. This CG position is evaluated for several loading cases, which include variable payload and fuel, with engine and tail configurations taken into consideration as mentioned in the previous chapter. To evaluate forward aircraft CG position, three cases for payload variation are implemented which are: empty aircraft, full payload, and simplified forward window-seating rule. Each case with and without fuel is developed. Aft CG position is implemented by considering two payload cases which are: maximum takeoff weight with 20% payload at the rear of the cabin (rear luggage hold) and simplified rear window-seating rule. Again with and without fuel is developed. A sample result for Boeing 737-800 aircraft is shown in **Figure 4-8**.

```
C.G. Position From Nose Apex.
Empty aircraft          (m) = 17.66
Datum Position -- 50%fuel full payload  (m) =16.77
Aft Limit               (m) = 17.94
Forward Limit           (m) = 16.76
```

Figure 4-8: Sample CG output result

4.3.4 Aerodynamics

This module computes the aerodynamic coefficients required for the designed aircraft which are: zero-lift drag, aircraft lift, lift induced drag, total aircraft drag, flap effects, and stall speeds. Zero-lift drag and lift induced drag are implemented by summing the effects from each aircraft component, plus the effect of the interference at the wing/fuselage junction. Aircraft lift is implemented via the usual way which is the ‘Aircraft-less-tail’ plus the tail effects. Total drag is the summation of zero-lift drag and lift induced drag. **Figure 4-9** shows a sample of a lift-drag polar plot for Boeing 737-800 aircraft.

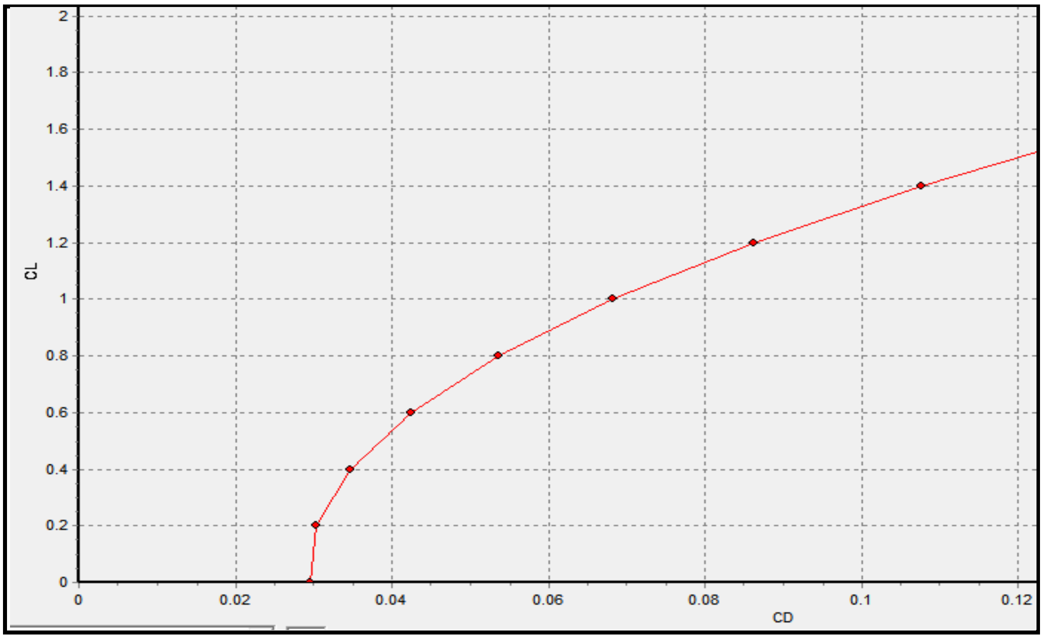

Figure 4-9: Lift-Drag polar plot

Flap effects are also included in evaluating aircraft lift at take-off and landing based on Torenbeek's formulae. Stall speeds for takeoff and landing are implemented as well. **Figure 4-10** shows the results of this module.

4.3.5 Stability

Implementation of this module is necessary to calculate the static stability represented by the static margin. Evaluation of the static margin is based on both the aft CG position (which is calculated in the CG module) and the position of the aircraft neutral point (which is calculated in this module). **Figure 4-10** shows these results for Boeing 737-800 aircraft.

```
** Aerodynamics Data ***
*** Zero-Lift Drag Coefficients ***
Exposed Wing     = 0.006412
Fuselage         = 0.008074
Nacelles(total)  = 0.010182
Horizontal Tail  = 0.002004
Vertical Tail    = 0.001359
Interference     = 0.000590
&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&&

      CL          CD
     ----       ---------
     0.00       0.029477
     0.20       0.030406
     0.40       0.034742
     0.60       0.042485
     0.80       0.053635
     1.00       0.068192
     1.20       0.086157
     1.40       0.107528
     1.60       0.132307

Neutral Point (POWER-OFF)         (m) = 18.49 from
Nose
Static Margin (DATUM C.G., POWER-OFF) =  0.42

         Flap Defln.  Section CL   Wing CL  Trimmed
a/c CL
            (deg)         (max)        (max)        (max)
TakeOff =   24.70         3.25         2.40         2.31
Landing =   40.00         3.50         2.56         2.45
```

Figure 4-10: Sample results of aerodynamics and stability modules

4.3.6 Flight performance

This module is implemented for both the flight profile analysis and field performance. Three phases of the flight profile (climb, cruise, and descent) are analysed separately and summed to determine the flight performance parameters. Time and fuel are also calculated using linear interpolation in each stage. The engines are at the maximum continuous climb rating during the climb phase while the descent phase analysis is performed at flight idle setting. **Figure 4-11** shows the output of the flight profile analysis for Boeing 737-800 aircraft. Additionally, three major parameters are computed which are: Balanced Field Length (BFL), Landing Field Length (LFL), and Second segment climb gradient (both ISA and WAT shown as **Figure 4-12** for Boeing 737-800 aircraft.

```
********* First stage ****************
Initial mass (kg) =  83192.3
 ----------------- climb ------ cruise ----- descent
---
Distance   (m) = 156704.2    5041850.0     141182.2
Fuel burn (kg) =   1152.7      15066.3          0.0
Time       (s) =    958.6      21009.4        790.8
IAS      (m/s) =    119.57       141.93       139.21

Cruise Altitude        (m) =  9690
Cruise Thrust Setting (%) = 86.00

*********** R. O. D / C ************
Start of climb    (m/s) = 14.67
End of climb      (m/s) =  6.11
Start of descent (m/s) = -14.10
End of descent    (m/s) = -10.64

********* Diversion stage ****************
Initial mass (kg) =  66928.0
----------------- climb ------ cruise ----- descent
---
Distance   (m) =  47223.8     49950.0      84891.2
Fuel burn (kg) =    470.1       162.1          0.0
Time       (s) =    333.8       389.7        523.9
IAS      (m/s) =    119.57       93.53       139.21

Cruise Altitude        (m) =  6096
Cruise Thrust Setting (%) = 38.00

*****************R. O. D / C ************
Start of descent (m/s) = -12.78
End of descent    (m/s) = -10.60
```

Figure 4-11: Flight profile performance output results

```
============ Field Performance ==============
Second segment gradient    = 0.008
Balanced field length (m) = 2640
Takeoff stall speed (m/s) =  68.0
Landing mass           (kg) =  66928
Landing field length   (m) = 1371
Landing stall speed (m/s) =  59.2

+++++++++++ WAT Performance ++++++++++++++++
At ISA + 20 deg.elevation (m) =      0
Second segment climb gradient = 0.007
```

Figure 4-12: Flight field performance output results

4.3.7 Cost estimation

Finally, the cost estimation module computes the aircraft cost and direct operating cost (DOC). DOC is a significant parameter of interest when it comes to commercial aircraft design. AEA's method is a standard method used in this module. **Figure 4-13** shows the detail output of cost estimation module for Boeing 737-800 aircraft.

```
===== Cost Estimation (Dollars 2010) ========
Stage length               (km) =  5665
Fuel price              ($/USG) = 2.15
Fuel used                  (kg) = 16264
Block time              (hours) =  6.44
Price of airframe          ($M) = 72.92
Price of engines           ($M) = 10.89
Price of aircraft          ($M) = 83.81

====== Cost/Flight ----- (AEA method) ===========
Utilization             (hours/year) =  3489
Depreciation cost                ($) = 11602.2
Insurance cost                   ($) =   801.4
Interest cost                    ($) =  9565.4
Standing charge (Dep+Ins)        ($) = 12403.6
Standing charge (Dep+Ins+Int)    ($) = 21969.0
Total aircraft maintance         ($) =  1760.9
Fuel & oil cost                  ($) = 11347.2
Flight Crew cost                 ($) =  5784.5
Cabin Crew cost                  ($) =  2702.1
Landing Fee                      ($) =  1141.1
Navigation Fee                   ($) =  6424.9
Ground-Handling charges          ($) =  3995.0

--------------------- With Interest  Without Interest
##############################################
Total DOC/flight   ($) = 55124.8        45559.5
Total DOC/mile  ($/nm) =    18.0           14.9
Seat mile cost  (c/nm) =     9.534          7.9
```

Figure 4-13: Output results of cost estimation module

4.4 Parametric studies

An important aspect, that aids understanding and explains the philosophy of aircraft design are the parametric studies. In conducting a parametric study, the program executes the synthesis program many times with different sets of design variables. As the design enters a new phase, the student can assess the impact of changing certain parameters in the design. Student can assign parameters for evaluation, define the parameter range, specify the design goal (output parameter), and analyse the results of each parameter variation. It performs so-called "*What if* scenarios" which means: *what* happens *if* one or more design parameters are changed. Implementing this module requires the following:

1. Selection of the design variables.
2. Range of design variables.
3. Selection of the dependent variable.

Each section in the synthesis program has design variables associated with it. One approach to working with design variables is to create a user-defined list of parameters for a specific area of interest. Since the students at this stage may not have a full appreciation of the impact and influence of the design variables, a user selectable list may not serve the intended purpose. The other method is a pre-defined list of variables, presented as a drop-down list, that are considered to be important parameters and have a great impact on the final aircraft design. For example, in the wing section, aspect ratio and wing area have a great impact on aircraft weight and stability. Hence both are part of the pre-configured list.

The design variables range means the boundaries between which the design variable value is allowed to be changed. Lower and upper boundaries represent the lowest and highest value respectively that the design variable should have. The common way to implement these boundaries is as a percentage of the nominal design variable value. A symmetric percentage is the simplest approach for implementation. For example, 20% means that the variable value starts with -20% (low boundary) and moves toward +20% (high boundary) in a pre-defined incremental sequence.

There are many output or calculated parameters, only the most significant parameters are presented to the user for selection. The output results of these studies are presented as graphs. Graphical representation of information is easy to understand and aids appreciation of the effect of the parameter variation. Colours are used to differentiate between plots in a single graph which can be viewed by the student as 2D or 3D plots.

Another feature has been added which allows the selection of two independent design variables simultaneously, with one output variable. The primary design variable is represented by the bottom axis (x-axis) while the secondary variable is represented by the right axis. **Figure 4-14** shows the output form of this module. The primary design variable is the Wing area and the output is the MTOW. The multiple selection of an independent variable allows the examination of changing aspect ratio at the same time.

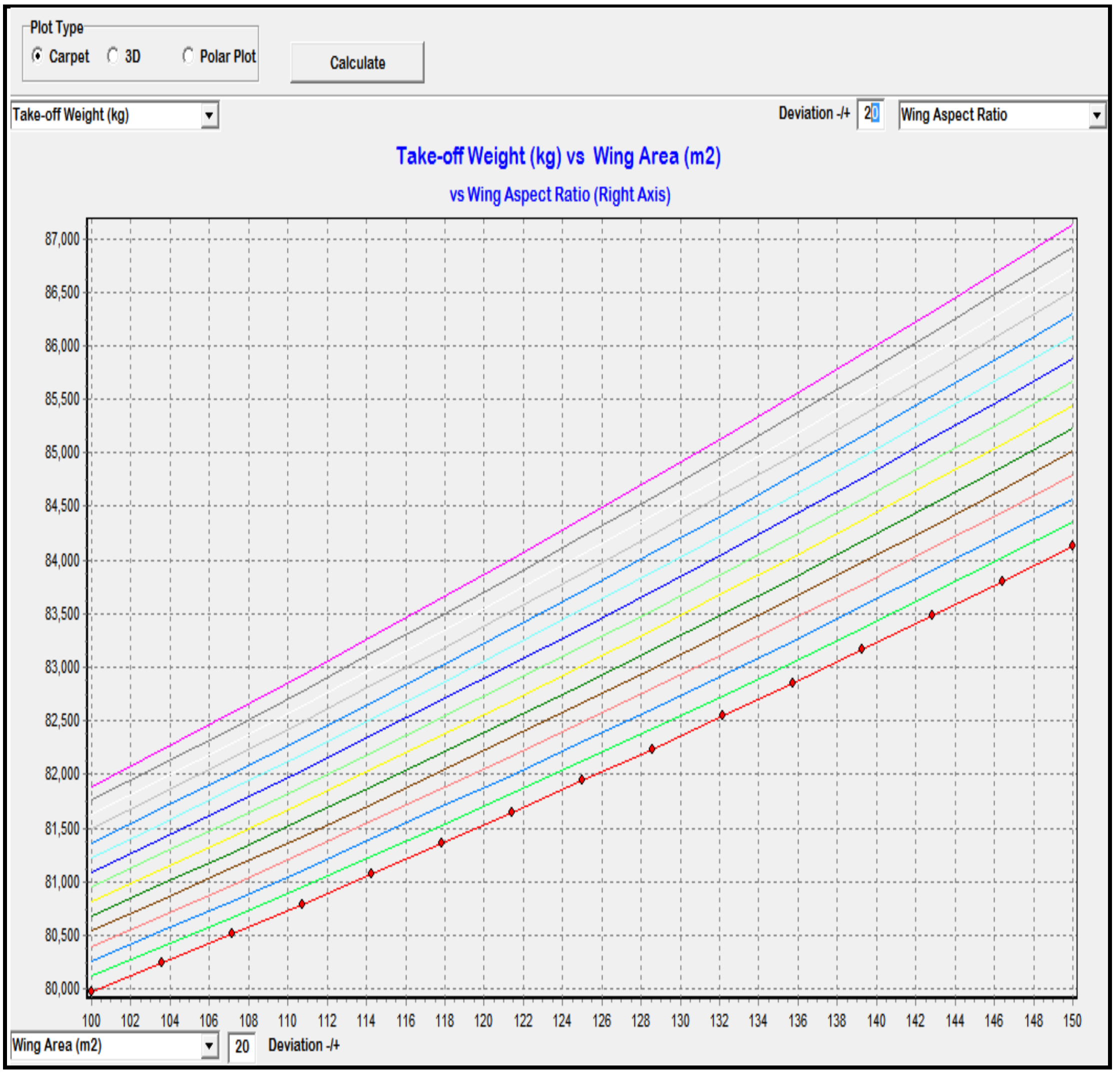

Figure 4-14: Parametric studies output form

4.5 Takeoff

Although balanced field length (BFL) is calculated in the performance module of the synthesis program, an additional module is employed to synthesise the takeoff stage in much more detail. Takeoff definition under FAR 25 is "*an aircraft taking off performs a ground roll to rotation velocity, rotates to liftoff attitude, lifts off and climbs to a height of 35 ft*". Two types of takeoff are recognised which are: takeoff with all engines operating (AEO) and takeoff with one engine inoperative (OEI). The aircraft accelerates from a stop or taxi speed to the velocity of rotation (V_r) rotates to the liftoff attitude with corresponding velocity (V_{lo}) and climbs over an obstacle of 35 feet. The takeoff safety speed, which is given the designation (V_2), is the velocity at the end of the 35 ft climb. Two methods have been developed to analyse the equation of motion during the takeoff stage. The first method was proposed by Powers [102] and called simplified Powers method. This method has no thrust vectoring capability, since the assumption was made that the aircraft thrust is constant during the takeoff run. Also, one equation of motion is used during the aircraft takeoff. The simplified Powers method has several disadvantages. First of all, the method produces surprisingly good results for conventional aircraft, but aircraft with large high lift devices or with STOL capability will not produce good results. Secondly, it does not allow STOL or unconventional high lift devices to be properly modelled due to its small amount of input data. Finally, the absence of a rotation phase has proven to under-predict ground runs in many situations. This has proven to be more of a problem in aircraft with high thrust to weight ratios (T/W) and in aircraft with especially high wing loadings (W/S).

The second method was proposed by Krenkel & Salzman [103]. This method calculates the aircraft equations of motion for ground roll and climb. It has vectored thrust as the assumption is made that the thrust varied with velocity and it did not solve for balance field length. In this method, the ground roll equations were obtained by summing the forces on the aircraft in the horizontal direction and substituting them into Newton's second law ($\sum F=ma$). This caused the algorithm to consistently under predict most aircraft takeoff times, distances and velocities. The modified approach to this method [104] is based on time-step integration technique, while the original method solved the aircraft equations of motion parametrically. It is also modified for calculating BFL for preliminary design purposes. *"Four equations were developed for the climb phase. Two are the governing balance of force equations, one relates the* *velocity components to the climb angle, γ, and the fourth relates the total velocity, V, to the individual velocity components. The result is a system of nonlinear ordinary differential equations. The ground and climb equations of motion, with quadratic thrust variation with velocity and a rotation phase included, were used to solve for both AEO and balanced field length OEI takeoff."* [104]. Many advantages for this method over the simplified Powers method have been noted. The first advantage is the quadratic thrust variation with velocity allows a more accurate engine representation. The second advantage is the ability to specify aerodynamic parameters during ground run and during climb. This allows the user much more control over the aircraft takeoff as well as being a better representation of true takeoff. The third advantage is the ability to vector the thrust during the takeoff run, allows application of this methodology to STOL aircraft. The last advantage is, like the simplified Powers' method, the modified Krenkel and Salzman method requires little data to produce results. Therefore, it is recommended for use in the preliminary design phase due to its accuracy in predicting the takeoff parameters.

This module is an implementation of the modified Krenkel and Salzman method. The module program analyses the takeoff stage for the two cases; normal takeoff (all engines operating), and BFL (one engine inoperative). In each case, different aspects in takeoff analysis are described. **Figure 4-15** shows the output results for both cases, i.e. AEO and OEI, which include the important velocities, distances, and times. As mentioned above, normal takeoff and BFL calculations are iterative. The number of iterations for BFL calculations is also shown in the figure which is dependent on the prescribed error tolerance to balance between the run time and the accuracy of calculations. Convergence in less than 20 iterations is indicative of numerical stability in the integration process and the pre-set convergence tolerance. A tighter tolerance will require many iterations before the convergence criterion is satisfied. Data presented in **Figure 4-15** pertains to the take-off performance of the Boeing 737-800 aircraft. The computed values in terms of various speeds and distances compare favourably with the published results from Reference [105]. For instance the quoted value for the rotation velocity of the 737-800 at an MTOW of 82 tonnes, dry runway, zero headwind is 97 m/s where as the program computes the same value as 87.09 m/s. This small variation is due to the differences in the aerodynamic parameters used for the ground run and the friction characteristics of the runway. The method is sufficiently accurate for use in the preliminary design phase. A similar pattern is observed for other parameters.

```
NORMAL TAKEOFF SUMMARY
----------------------
Rotation Velocity    (Vr):=     87.090(m/s)
Liftoff Velocity    (Vlo):=     93.791(m/s)
Velocity over obs.(Vobs):=     98.274(m/s)
Rotation Distance    (Xr):=   1596.890(m)
Liftoff Distance    (Xlo):=   1868.232(m)
Distance to obst   (Xobs):=   2100.873(m)
Rotation Time        (Tr):=     35.660(s)
Liftoff Time        (Tlo):=     38.660(s)
Time to obst       (Tobs):=     41.090(s)

TOTAL TAKEOFF DIST (Xto):=   2100.873(m)
TOTAL TAKEOFF TIME (Tto):=     41.090(s)

 Iteration ....1
 Iteration ....2
 Iteration ....3
 Iteration ....4
 Iteration ....5
 Iteration ....6
 Iteration ....7
 Iteration ....8

OEI TAKEOFF SUMMARY
-------------------
Critical Velocity     (Vcrit)  :=     77.266(m/s)
Decision Velocity        (V1)  :=     80.144(m/s)
Velocity over obs      (Vobs)  :=     98.274(m/s)
Critical Distance     (Xcrit)  :=   1240.708(m)
Decision Distance        (X1)  :=   1476.832(m)
Balanced Field Length (BFL1)   :=   2641.844(m)
Critical Time         (Tcrit)  :=     31.329(s)
Decision Time            (T1)  :=     34.329(s)
OEI Takeoff Time       (TBFL)  :=     62.828(s)

-END of COMPUTATION -
```

Figure 4-15: Takeoff analysis output results

4.6 Optimisation

As mentioned in chapter two, numerical optimisation is one feature that is an essential requirement for preliminary design software for teaching. Optimisation methods can be classified into two categories; classical and advanced techniques [106].

1. Classical Techniques: these are useful in finding the optimum solution or unconstrained maxima or minima of continuous and differentiable functions. The classical methods are analytical methods and make use of differential calculus in locating the optimum solution. These methods have limited scope in

practical applications as some of them involve objective functions, which are not continuous and/or differentiable, and lead to a set of nonlinear simultaneous equations that may be difficult to solve.

2. Advanced Techniques: these are classified into four methods; hill climbing, simulated annealing, genetic algorithms, and ant colony.
    i. Hill climbing: in this method, the first closer node is chosen whereas in steepest ascent hill climbing**,** all successors are compared and the closest to the solution is chosen. Both forms fail if there is no closer node. This may happen if there are local maxima in the search space which are not solutions. This method is widely used in artificial intelligence fields, for reaching a goal state from a starting node.
    ii. Simulated annealing: this method comes from annealing process in metallurgy. It is a technique involving heating and controlled cooling of a material to increase the size of its crystals and reduce their defects. In this method, each point of the search space is compared to a state of some physical system, and the function to be minimised is interpreted as the internal energy of the system in that state. Therefore, the goal is to bring the system, from an arbitrary initial state, to a state with the minimum possible energy.
    iii. Genetic algorithms: this method is a local search technique used to find approximate solutions to optimisation and search problems. It is a particular class of evolutionary algorithm that use techniques inspired by evolutionary biology such as inheritance, mutation, selection, and crossover (it is also called recombination). This method is typically implemented as a computer simulation, in which a population of abstract representations (called chromosomes) of candidate solutions (called individuals) to an optimisation' problem, evolves toward better solutions.
    iv. Ant colony: the idea of this method comes from the real world. Ants (initially) wander randomly, and upon finding food return to their colony while laying down pheromone trails. If other ants find such a path, they are likely not to keep travelling at random, but instead follow the trail laid by earlier ants, returning and reinforcing it if they eventually find

food. The aim of this algorithm is to mimic this behaviour with "simulated ants" walking around the search space representing the problem to be solved. It is used to produce near-optimal solutions to the travelling salesman problem. Ant colony algorithm has an advantage over simulated annealing and genetic algorithm approaches when the graph may change dynamically. It can be run continuously and can adapt to changes in real time. Ant colony method is of interest in network routing and urban transportation systems.

An optimiser (RQPMIN) from RAE [107] has been added to allow a greater flexibility in selecting objective functions and in exploring the optimised design to change in some specifications or constraints. It is a general numerical optimisation program that can be applied in a wide range of situations for the solution of many problems in science, engineering, and commerce, involving the calculation of the optimum value. The optimiser has the ability to handle problems with up to 75 constraints and up to 50 variables. It was written in standard Fortran 77 and re-encoded in Delphi to be suitable for use with the software.

The optimiser algorithm uses ‘hill-climbing’ method which is very fast with respect to other methods. It is of interest for interactive software and Problem-Based Learning purposes. It is based on the Lagrange-Newton approach, i.e. a stationary point of the Lagrange function is calculated by Newton’s method. The concept of pseudo-feasibility is used where a trial point is rejected if the square root of the sum of the squares of the constraints is greater than the radius of pseudo-feasibility. The latter is initially given a substantial value, being only reduced when necessary [108][109].

The input data consists of three groups:

1. List of variables.
2. List of constraints.
3. Objective function.

A list of variables were selected from the input design variables of the synthesis program and finalised during the development of the software. The selection process was based on RAE requirements for compatibility with the swept synthesis. Changes in the value of these variables may change the overall design of the aircraft. Twentyone design variables were selected and shown in **Figure 4-16**. Another benefit of the optimiser flexibility is that the student can decide which of the design variables are to be freed or fixed. By varying the status of these variables, in-depth studies can be conducted by students to enhance their understanding and analysis of aircraft design process.

The list of constraints comprises fourteen equality or in-equality constraints which include fuel weight, total takeoff weight, balanced and landing field lengths, static margin, stage length, etc) which can be set by the students. **Figure 4-17** shows the form that allows the assignment of these constraints.

The objective function represents the minimum value of the function required for an optimal solution. Main objective functions include, total fuel weight, takeoff weight, direct operating cost, and wing weight. These functions are available for the student to select as shown in **Figure 4-18**.

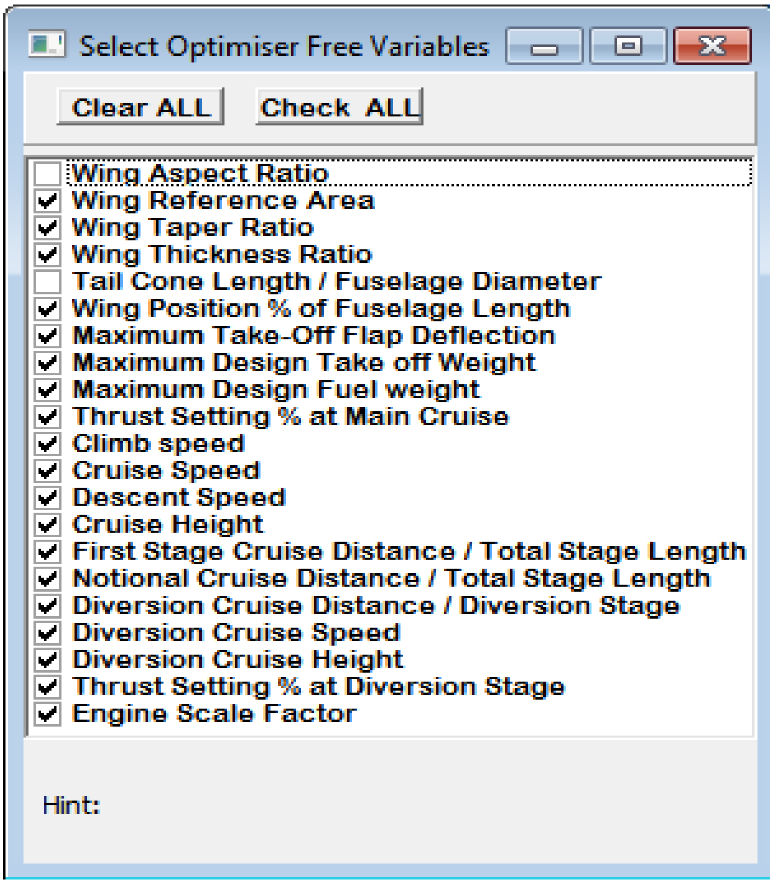

Figure 4-16: Input design variables form of the optimiser

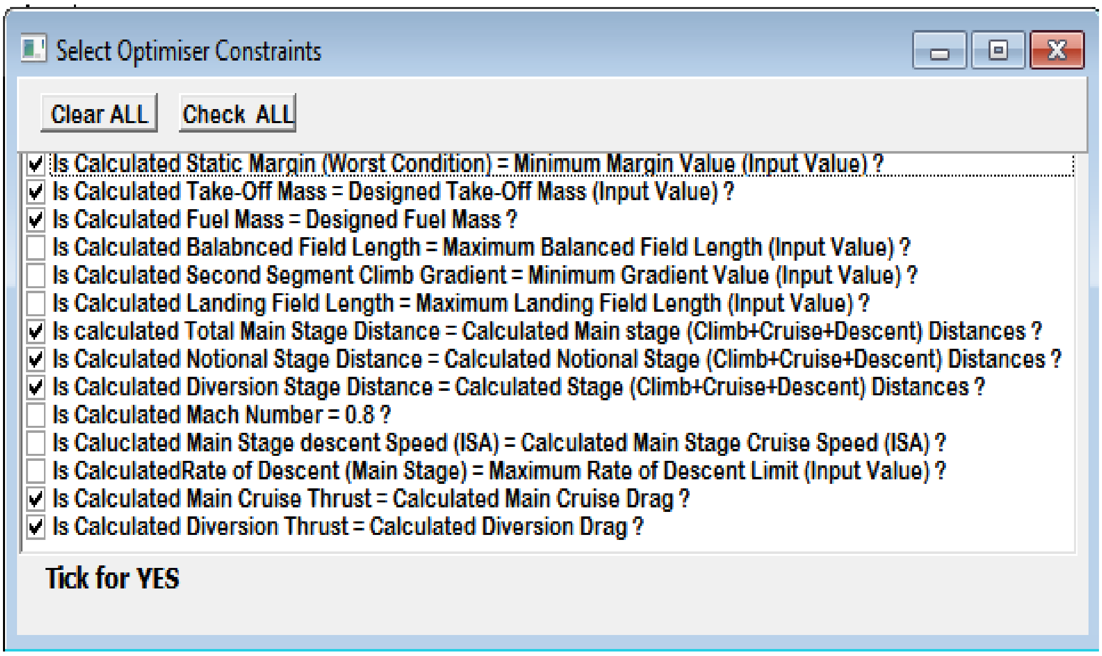

Figure 4-17: Constraints form of the optimiser

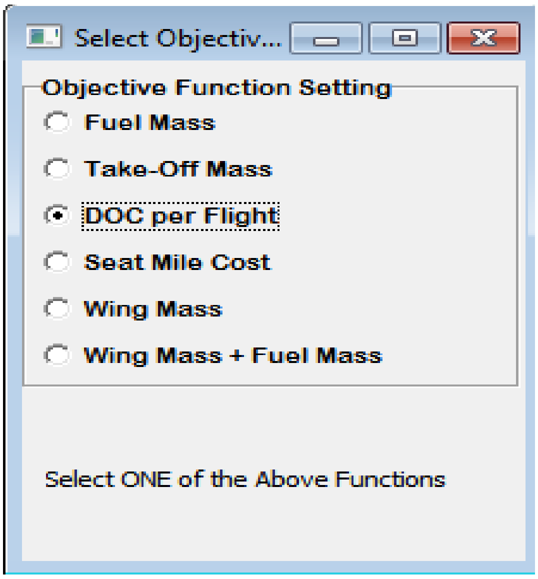

Figure 4-18: Objective function form of the optimiser

The operation of the optimiser starts by checking of the input data and if no errors are detected it proceeds to program initialisation. The optimiser calls the synthesis program a few times and evaluates the objective and constraint functions using the initial values of the design variables. Then, it begins a series of feasibility steps in order to locate a feasible path from which it can start minimising the objective function. During this stage, some values of the design variables may change. The optimiser will stop and notify the student that no further progress is possible if no feasible path has been located. After the feasible path has located, the optimiser begins a long series of minimisation steps which continues until the objective and constraint function values minimise within the prescribed tolerances and convergence is detected. Three situations of convergence (A, B, and C) are classified [109]:

1. *Convergence A: occurs when the optimiser has located a point at which the estimated distances to the minimum of the criterion function and to the minimum of the sum of the squares of the constraint function are less than the prescribed tolerances.*
2. *Convergence B: occurs when the optimiser has located a point at which the estimated distance to the minimum of the sum of the squares of the constraint functions is less than the prescribed tolerance and a value of the criterion function sufficiently smaller than the current value cannot be found.*
3. *Convergence C: occurs when the optimiser has located a point at which the estimated distance to the minimum of the sum of the squares of the constraint function is less than the prescribed tolerance where no further changes to the design variables can be made.*

4.7 Dynamic stability

The design of an aircraft must meet stability requirements to be able to fly. "*An understanding of flight stability and control played an important role in the ultimate success of the earliest aircraft designs.*" [90]. It is essential for students to evaluate both static and dynamic stability concepts early in the preliminary stage. This early evaluation, before the detail design phase takes place, makes the design process more efficient and speeds up the design process itself.

It should be noted that the symbols and notation used in this section that pertain to the aspects of evaluating the dynamic stability of the design are not noted in the list of symbols. Instead the appropriate symbol is defined in the text on its first use.

Figure 4-19A (a) depicts an aircraft in steady flight. In this condition sum of all forces and moments is zero. An aircraft would be termed to be statically unstable if owing to an applied moment, the aircraft does not return to its initial undisturbed state, **Figure 4-19A (b)**. However, if the aircraft on application of some moment, assumes a new state then it is termed to be statically neutral.

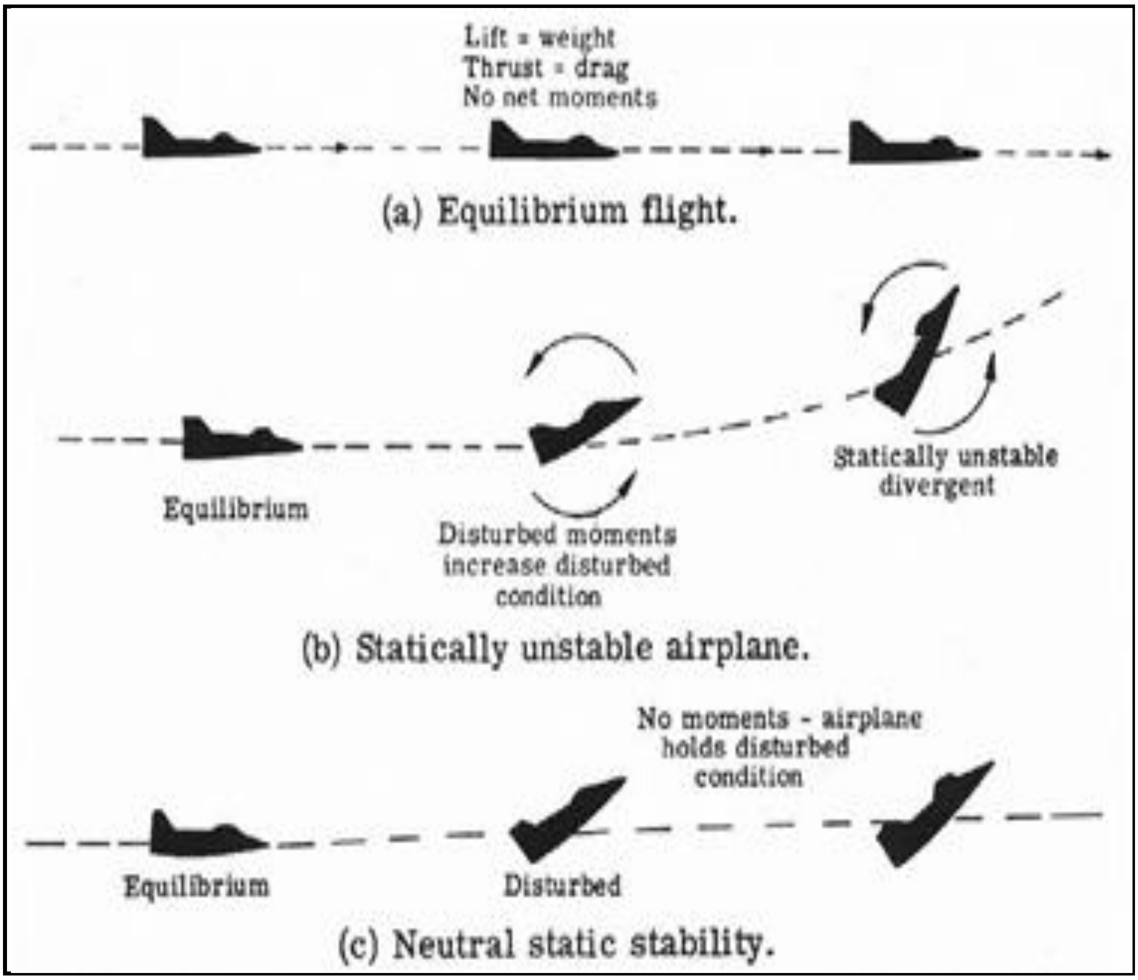

Figure 4-19A: Equilibrium flight and static stability

If an aircraft owing to some disturbance returns to its trimmed condition and if all dynamic oscillations decay in time then it statically and dynamically stable **Figure 4-19B (a)**. If however, the oscillations persist with equal amplitude the aircraft possesses neutral dynamic stability, **Figure 4-19B (b)**. And when the amplitude of oscillations increases with time then it is termed dynamically unstable **Figure 4-19B (c)**.

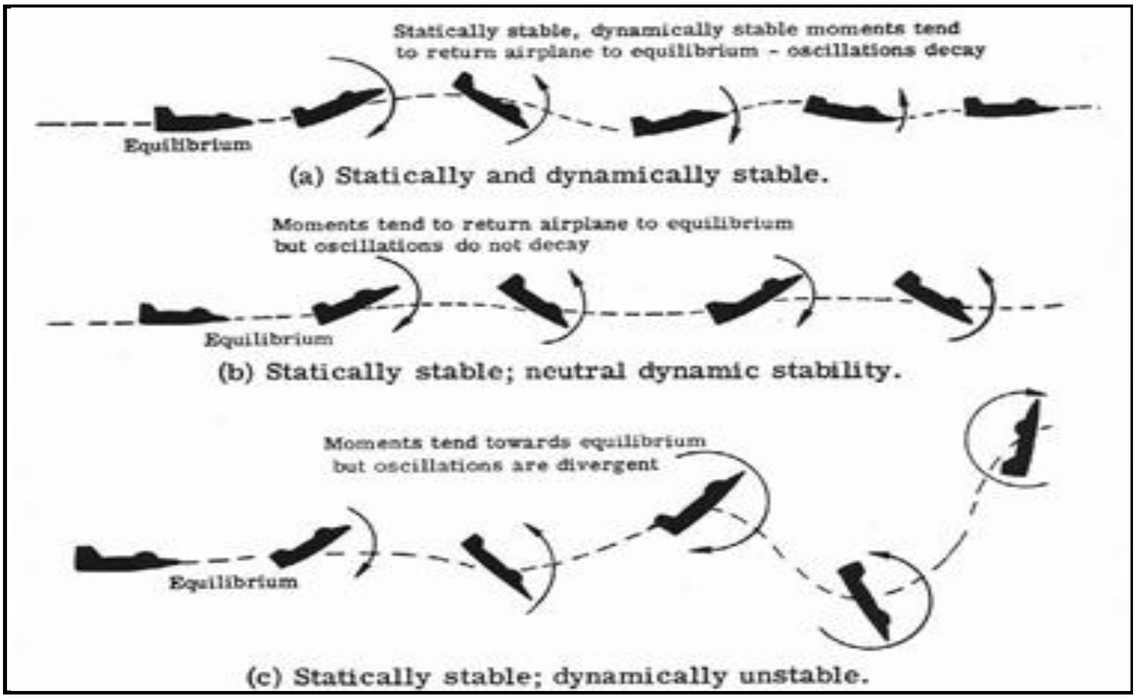

Figure 4-19B: Static and dynamic stability

Static stability is defined as the initial tendency of an aircraft to return to its equilibrium (original) state after it is disturbed (i.e. positive stability). Equilibrium is the original state for the aircraft which means trimmed flight in which sum of forces and moments are zero. If an aircraft initially tends to return to its trim flight condition after a disturbance, then it is statically stable. If it initially tends to move away, then it is statically unstable. Evaluation of static stability is achieved by looking at the aircraft stability in one direction at a time, i.e. longitudinal (pitch), lateral (roll), and directional (yaw) stability. If the pitching moment coefficient value is negative ($C_{m_\alpha} < 0$), then the aircraft will have longitudinal static stability. If the rolling moment coefficient value is negative ($C_{l_\beta} < 0$), then the aircraft will have lateral static stability. If the yawing moment coefficient value is positive ($C_{n_\beta} > 0$), then the aircraft will have directional static stability.

The neutral stability means that an aircraft tends to stay in its most recently commanded attitude or condition, without oscillations, and it will neither tend to return to its previous state or diverge from its new attitude. It is also called *neutral equilibrium* or *neutral static stability*. It would exhibit neutral stability only if the centre of pressure and centre of gravity coincide.

On the other hand, dynamic stability refers to how an aircraft moves after it is disturbed from a trim condition. Evaluation of dynamic stability is done in a similar way to that for static stability except roll and yaw motions are now coupled (which means that one does not happen without the other). In general, there are two types of aircraft motion: longitudinal and lateral-directional motion. Longitudinal (pitch) motion consists of two distinct oscillation modes. The primary mode is called short-period mode which is a heavily damped oscillation. It is a decaying oscillation that occurs in the aircraft's angle of attack when the aircraft pitches up or down. In aircraft design, short-period mode should have high frequency and should decay quickly. Therefore, decreasing C_{m_α} increases the frequency of the short-period mode, i.e. improves its stability. The other mode is called long-period mode ("phugoid" mode) which is a lightly damped oscillation and is a large-amplitude variation in air-speed and altitude. The motion is so slow that the effects of inertial forces and damping forces are very slow. Typically, the period is 20-60 seconds where the pilot generally can control this situation easily. In lateral-directional motion, two types of motion occur: spiral mode and Dutch roll. Improving lateral-directional stability can be achieved by reducing the impact of both spiral mod and Dutch roll. Decreasing rolling moment coefficient C_{l_β} improves the spiral mode (more stable), but worsens the Dutch roll (increasing the amplitude of the Dutch roll), while increasing yawing moment coefficient C_{n_β} improves the Dutch roll (increases the frequency of the Dutch roll), but worsens the spiral mode (i.e. less stable).

Evaluation of dynamic stability coefficients is a difficult challenge for aircraft designers. The difficulty increases in the early stages of the aircraft design process due to the uncertainty of detailed information about the design. The traditional method is the wind tunnel test which has been used for a long time. It requires both the construction of a model and an adequate test facility. Also, the lag time between the paper design and test results can be significant. Furthermore, any change in aircraft configuration requires a change of the test model. In the preliminary design phase, the alternative approach involves: empirical methods such as USAF DATCOM [110][111], numerical methods using computational fluid dynamics (CFD) [112][113], and estimation from a basic stability theory method [9], are used. Each of these approaches has its own advantages and disadvantages with respect to the range of applicability, computation time, and cost [114]. For undergraduate teaching purposes, estimation of dynamic stability coefficients based on basic stability theory is used for implementing the dynamic stability module. *Simple theory is only accurate for preliminary design or relationships between the overall aircraft geometry and stability* [115].

Table 4-1 and **Table 4-2**, [90], represent the simplified equations to predict the most important longitudinal and lateral stability coefficients (non-dimensional derivatives). These derivatives were derived from the nonlinear differential equations of aircraft motion. Small-perturbation theory was used to linearise these equations [116]. Equations to predict directional (or dimensional) longitudinal and lateral derivatives are shown in **Table 4-3** and **Table 4-4**, [90], respectively. Rearrangement of the directional equations of motion (both longitudinal and lateral) into state space forms is useful from control standpoint, as shown in **Figure 4-20** and **Figure 4-21**, [90]. The state space model can therefore be used in any further control scheme easily, by using tools such as MATLAB [117]:

$$\dot{x} = Ax + Bu \tag{4.1}$$

Where A is $n \times n$ matrix, that consists of the stability derivatives and B is $n \times m$ control driving matrix, that contains control derivatives, and x is a $n \times 1$ state vector that consists of the principal motions of the aircraft, and may have linear or rotational accelerations, rates or positions.

The eigenvalues of the A matrix, are also the roots of the conventional characteristic equation. If the roots or eigenvalues are negative or have negative real parts then the system has stability, positive real parts of roots are indicative of instability. The roots of the system can also be evaluated by using the characteristic equation viz.:

$$det(\lambda I - A) = 0 \tag{4.2}$$

For longitudinal stability, two modes are distinct as mentioned before. **Table 4-5**, [90], shows the approximate relationships for the long- and short-period modes. These relationships were developed by assuming that the long-period mode occurred at a constant angle of attack and the short-period mode occurred at a constant speed. "*These assumptions were verified by an examination of the exact solution. The* *approximate formulas permitted us to examine the relationship of the stability derivatives on the longitudinal motion."* [90].

Equations for estimating the longitudinal stability coefficients

	X-force derivatives	Z-force derivatives	Pitching moment derivatives
u	$C_{X_u} = -[C_{D_u} + 2C_{D0}] + C_{T_u}$	$C_{z_u} = -\frac{M^2}{1-M^2}C_{L0} - 2C_{L0}$	$C_{m_u} = \frac{\partial C_m}{\partial M}M_0$
α	$C_{X_\alpha} = C_{L0} - \frac{2C_{L0}}{\pi e}\frac{C_{L\alpha}}{AR}$	$C_{Z_\alpha} = -(C_{L\alpha} + C_{D0})$	$C_{m\alpha} = C_{L\alpha w}\left(\frac{X_{cg}}{\bar{c}} - \frac{X_{ac}}{\bar{c}}\right) + C_{m\alpha\,\mathrm{fus}} - \eta V_H C_{L\alpha_t}\left(1 - \frac{d\varepsilon}{d\alpha}\right)$
$\dot{\alpha}$	0	$C_{Z\dot{\alpha}} = -2\eta C_{L\alpha_t} V_H \frac{d\varepsilon}{d\alpha}$	$C_{m\dot{\alpha}} = -2\eta C_{L\alpha_t} V_H \frac{l_t}{c}\frac{d\varepsilon}{d\alpha}$
q	0	$C_{z_q} = -2\eta C_{L\alpha_t} V_H$	$C_{m_q} = -2\eta C_{L\alpha_t} V_H \frac{l_t}{c}$
α_e	0	$C_{Z\delta_e} = -C_{L\delta_e} = -\frac{S_t}{S}\eta\frac{dC_{L_t}}{d\delta_e}$	$C_{m\delta_e} = -\eta V_H \frac{dC_{L_t}}{d\delta_e}$

Subscript 0 indicates reference values and M is the Mach number.

AR	Aspect ratio	V_H	Horizontal tail volume ratio
C_{D0}	Reference drag coefficient	M	Flight mach number
$C_{L\alpha}$	Reference lift coefficient	S	Wing area
$C_{L\alpha}$	Airplane lift curve slope	S_t	Horizontal tail area
$C_{L\alpha w}$	Wing lift curve slope	$\frac{d\varepsilon}{d\alpha}$	Change in downwash due to a change in angle of attack
$C_{L\alpha_t}$	Tail lift curve slope	η	Efficiency factor of the horizontal tail
$\bar{c}$	Mean aerodynamic chord		
e	Oswald's span efficiency factor		
l_t	Distance from center of gravity to tail quarter chord		

Table 4-1: Non-dimensional longitudinal stability coefficients, **[90]**

Equations for estimating the lateral stability coefficients

	Y-force derivatives	Yawing moment derivatives	Rolling moment derivatives
β	$C_{y_\beta} = -\eta \frac{S_v}{S} C_{L_{\alpha_v}} \left(1 + \frac{d\sigma}{d\beta}\right)$	$C_{n_\beta} = C_{n_{\beta wf}} + \eta_v V_v C_{L_{\alpha_v}} \left(1 + \frac{d\sigma}{d\beta}\right)$	$C_{l_\beta} = \left(\frac{C_{l_\beta}}{\Gamma}\right)\Gamma + \Delta C_{l_\beta}$ (see Figure 3.11)
p	$C_{y_p} = C_L \frac{AR + \cos\Lambda}{AR + 4\cos\Lambda} \tan\Lambda$	$C_{n_p} = -\frac{C_L}{8}$	$C_{l_p} = -\frac{C_{L_\alpha}}{12} \frac{1 + 3\lambda}{1 + \lambda}$
r	$C_{y_r} = -2\left(\frac{l_v}{b}\right)(C_{y_\beta})_{\text{tail}}$	$C_{n_r} = -2\eta_v V_v \left(\frac{l_v}{b}\right) C_{L_{\alpha_v}}$	$C_{l_r} = \frac{C_L}{4} - 2\frac{l_v}{b}\frac{z_v}{b} C_{y\beta_{\text{tail}}}$
δ_a	0	$C_{n_{\delta_a}} = 2KC_{L0} C_{l_{\delta_a}}$ (see Figure 3.12)	$C_{l_{\delta_a}} = \frac{2C_{L_\alpha}\tau}{Sb} \int_{y_1}^{y_2} cy\, dy$
δ_r	$C_{y_{\delta_r}} = \frac{S_v}{S} \tau C_{L_{\alpha_v}}$	$C_{n_{\delta_r}} = -V_v \eta_v \tau C_{L_{\alpha_v}}$	$C_{l_{\delta_r}} = \frac{S_v}{S}\left(\frac{z_v}{b}\right) \tau C_{L_{\alpha_w}}$

AR	Aspect ratio	S	Wing area
b	Wingspan	S_v	Vertical tail area
C_{L0}	Reference lift coefficient	z_v	Distance from center of pressure of vertical tail to fuselage centerline
C_{L_α}	Airplane lift curve slope	Γ	Wing dihedral angle
$C_{L_{\alpha_w}}$	Wing lift curve slope	Λ	Wing sweep angle
$C_{L_{\alpha_t}}$	Tail lift curve slope	η_v	Efficiency factor of the vertical tail
$\bar{c}$	Mean aerodynamic chord	λ	Taper ratio (tip chord/root chord)
K	empirical factor	$\frac{d\sigma}{d\beta}$	Change in sidewash angle with a change in sideslip angle
l_v	Distance from center of gravity to vertical tail aerodynamic center		
V_v	Vertical tail volume ratio		

Table 4-2: Non-dimensional lateral stability coefficients, **[90]**

Summary of longitudinal derivatives

$$X_u = \frac{-(C_{D_u} + 2C_{D_0})QS}{mu_0}(\text{s}^{-1}) \qquad X_w = \frac{-(C_{D_\alpha} - C_{L_0})QS}{mu_0}(\text{s}^{-1})$$

$$Z_u = \frac{-(C_{L_u} + 2C_{L_0})QS}{mu_0}(\text{s}^{-1})$$

$$Z_w = \frac{-(C_{L_\alpha} + C_{D_0})QS}{mu_0}(\text{s}^{-1}) \qquad Z_{\dot{w}} = -C_{z\dot{\alpha}}\frac{c}{2u_0}QS/(u_0 m)$$

$$Z_\alpha = u_0 Z_{\dot{w}}(\text{ft/s}^2) \text{ or } (\text{m/s}^2) \qquad Z_{\dot{\alpha}} = u_0 Z_{\dot{w}} (\text{ft/s}) \text{ or } (\text{m/s})$$

$$Z_q = -C_{Z_q}\frac{c}{2u_0} QS/m \ (\text{ft/s}) \text{ or } (\text{m/s}) \qquad Z_{\delta_e} = -C_{Z\delta_e}QS/m \ (\text{ft/s}^2)$$

$$\text{M}_u = C_{m_u}\frac{(QSc)}{u_0 I_y}\left(\frac{1}{\text{ft}\cdot\text{s}}\right) \text{ or } \left(\frac{1}{\text{m}\cdot\text{s}}\right)$$

$$\text{M}_w = C_{m_\alpha}\frac{(QS\overline{c})}{u_0 I_y}\left(\frac{1}{\text{ft}\cdot\text{s}}\right) \text{ or } \left(\frac{1}{\text{m}\cdot\text{s}}\right) \qquad M_{\dot{w}} = C_{m\dot{\alpha}}\frac{\overline{c}}{2u_0}\frac{QS\overline{c}}{u_0 I_y}(\text{ft}^{-1})$$

$$\text{M}_\alpha = u_0\text{M}_w \ (\text{s}^{-2}) \qquad \text{M}_{\dot{\alpha}} = u_0\text{M}_{\dot{w}} \ (\text{s}^{-1})$$

$$\text{M}_q = C_{m_q}\frac{\overline{c}}{2u_0}(QS\overline{c})/I_y \ (\text{s}^{-1}) \qquad \text{M}_{\delta_e} = C_{m\delta_e}(QS\overline{c})/I_y \ (\text{s}^{-2})$$

Table 4-3: Dimensional longitudinal stability coefficients, **[90]**

Summary of lateral directional derivatives

$$Y_\beta = \frac{QSC_{y\beta}}{m}(\text{ft/s}^2) \text{ or } (\text{m/s}^2) \qquad N_\beta = \frac{QSbC_{n\beta}}{I_z}(\text{s}^{-2}) \qquad L_\beta = \frac{QSbC_{l\beta}}{I_x}(\text{s}^{-2})$$

$$Y_p = \frac{QSbC_{y_p}}{2mu_0}(\text{ft/s}) (\text{m/s}) \qquad N_p = \frac{QSb^2C_{n_p}}{2I_x u_0}(\text{s}^{-1}) \qquad L_p = \frac{QSb^2C_{l_p}}{2I_x u_0}(\text{s}^{-1})$$

$$Y_r = \frac{QSbC_{y_r}}{2mu_0}(\text{ft/s}) \text{ or } (\text{m/s}) \qquad N_r = \frac{QSb^2C_{n_r}}{2I_x u_0}(\text{s}^{-1}) \qquad L_r = \frac{QSb^2C_{l_r}}{2I_x u_0}(\text{s}^{-1})$$

$$Y_{\delta a} = \frac{QSC_{y\delta_a}}{m}(\text{ft/s}^2) \text{ or } (\text{m/s}^2) \qquad Y_{\delta_r} = \frac{QSC_{y\delta_r}}{m}(\text{ft/s}^2) \text{ or } (\text{m/s}^2)$$

$$N_{\delta_a} = \frac{QSbC_{n\delta_a}}{I_z}(\text{s}^{-2}) \qquad N_{\delta_r} = \frac{QSbC_{n\delta_r}}{I_z}(\text{s}^{-2})$$

$$L_{\delta_a} = \frac{QSbC_{l\delta_a}}{I_x}(\text{s}^{-2}) \qquad L_{\delta_r} = \frac{QSbC_{l\delta_r}}{I_x}(\text{s}^{-2})$$

Table 4-4: Dimensional lateral stability coefficients, **[90]**

$$
\mathbf{A} = \begin{bmatrix} X_u & X_w & 0 & -g \\ Z_u & Z_w & u_0 & 0 \\ M_u + M_{\dot{w}} Z_u & M_w + M_{\dot{w}} Z_w & M_q + M_{\dot{w}} u_0 & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}
$$

$$
\mathbf{B} = \begin{bmatrix} X_\delta & X_{\delta_T} \\ Z_\delta & X_{\delta_T} \\ M_\delta + M_{\dot{w}} Z_\delta & M_{\delta_T} + M_{\dot{w}} Z_{\delta_T} \\ 0 & 0 \end{bmatrix}
$$

Figure 4-20: Longitudinal state space matrices, [90]

$$
\begin{bmatrix} \Delta\dot{\beta} \\ \Delta\dot{p} \\ \Delta\dot{r} \\ \Delta\dot{\phi} \end{bmatrix} = \begin{bmatrix} \dfrac{Y_\beta}{u_0} & \dfrac{Y_p}{u_0} & -\left(1 - \dfrac{Y_r}{u_0}\right) & \dfrac{g \cos \theta_0}{u_0} \\ L_\beta & L_p & L_r & 0 \\ N_\beta & N_p & N_r & 0 \\ 0 & 1 & 0 & 0 \end{bmatrix} \begin{bmatrix} \Delta\beta \\ \Delta p \\ \Delta r \\ \Delta\phi \end{bmatrix} + \begin{bmatrix} 0 & \dfrac{Y_{\delta_r}}{u_0} \\ L_{\delta_a} & L_{\delta_r} \\ N_{\delta_a} & N_{\delta_r} \\ 0 & 0 \end{bmatrix} \begin{bmatrix} \Delta\delta_a \\ \Delta\delta_r \end{bmatrix}
$$

Figure 4-21: Lateral state space representation, [90]

Summary of longitudinal approximations

	Long period (phugoid)	Short period
Frequency	$\omega_{n_p} = \sqrt{\dfrac{-Z_u g}{u_0}}$	$\omega_{n\mathrm{sp}} = \sqrt{\dfrac{Z_\alpha M_q}{u_0} - M_\alpha}$
Damping ratio	$\zeta_p = \dfrac{-X_u}{2\omega_{n_p}}$	$\zeta_{\mathrm{sp}} = -\dfrac{M_q + M_{\dot{\alpha}} + \dfrac{Z_\alpha}{u_0}}{2\omega_{n\,\mathrm{sp}}}$

Table 4-5: Frequency and damping of the longitudinal modes, using 2-DOF approximation, **[90]**

For lateral stability, three types of modes are distinct:

1. Spiral mode characterises as a slowly convergent or divergent motion.
2. Roll mode, i.e. highly convergent motion.
3. Dutch roll mode, i.e. lightly damped oscillatory motion having a low frequency.

When Equation (4.2) is solved for the lateral case, five roots exist, one real and two as a complex conjugate pairs. The real root is associated with the spiral mode, whose value can be determined to be [90]:

$$\lambda_{spiral} = \frac{L_\beta \times N_r - L_r \times N_\beta}{L_\beta} \tag{4.3}$$

For a stable spiral, the nominator must be positive since L_β is usually a negative quantity.

The first complex root is associated with the roll mode which is highly damped, and can be approximated as a first order characteristic given by [59]:

$$\lambda_{roll} \cong -\frac{1}{\tau} \cong L_p \tag{4.4}$$

Where: τ is the roll time constant

The second complex root is associated with the Dutch Roll mode, the un-damped natural frequency and the damping ratio is given by the following:

$$\omega_{nDR} = \sqrt{\frac{Y_\beta \times N_r - N_\beta \times Y_r + u_0 \times N_\beta}{u_0}} \tag{4.5}$$

$$\zeta_{DR} = -\frac{1}{2 \times \omega_{nDR}} \times \left(\frac{Y_\beta + u_0 \times N_r}{u_0}\right) \tag{4.6}$$

In order to implement the dynamic stability module, two more significant input design variables are added. These are: velocity and height of the designed aircraft under consideration. Additional input values are required which are the initial state values for both longitudinal and lateral analysis. Output results are shown in a text format. This user interaction form has both inputs and outputs, and displays non-dimensional and dimensional derivatives, state space matrices and approximated modal values. Graphical outputs are also available to show the time response of each element in the state space representation. The system can be excited by one of two disturbance signals, impulse or step.

Figure 4-22 shows the dynamic stability module interaction form, for the Boeing 747-200 aircraft. For further analysis and the synthesis of feedback control, all the relevant data is exported as an '.m' file for use in MATLAB/SIMULINK software. Example outputs are presented in Appendix IV; the data applies to the case studies presented in chapter six.

Another feature was added to the dynamic stability module which is its ability to prepare the input file (which is called for005.dat) for use in the DATCOM software. This file defines the flight characteristics and geometry of the designed aircraft as explained in Appendix I. Due to its specific format, any mistake or deviation from the prescribed format will cause DATCOM software to raise an error. To avoid this problem, a GUI was developed to create the input file as shown in **Figure 4-23**.

Most of the input data are passed directly by the software developed as part of this research, while other data are entered manually through this interface. This feature allows the use of the DATCOM software for predicting the stability and control derivatives used in dynamic stability analysis.

The form is displayed as a tabbed notebook, and has multiple tabs. The additional variables required for DATCOM are input using this interface. The variables required are explained in Appendix I. These variables are grouped in fuselage, wing, empennage and aerofoil sections on the designed interface.

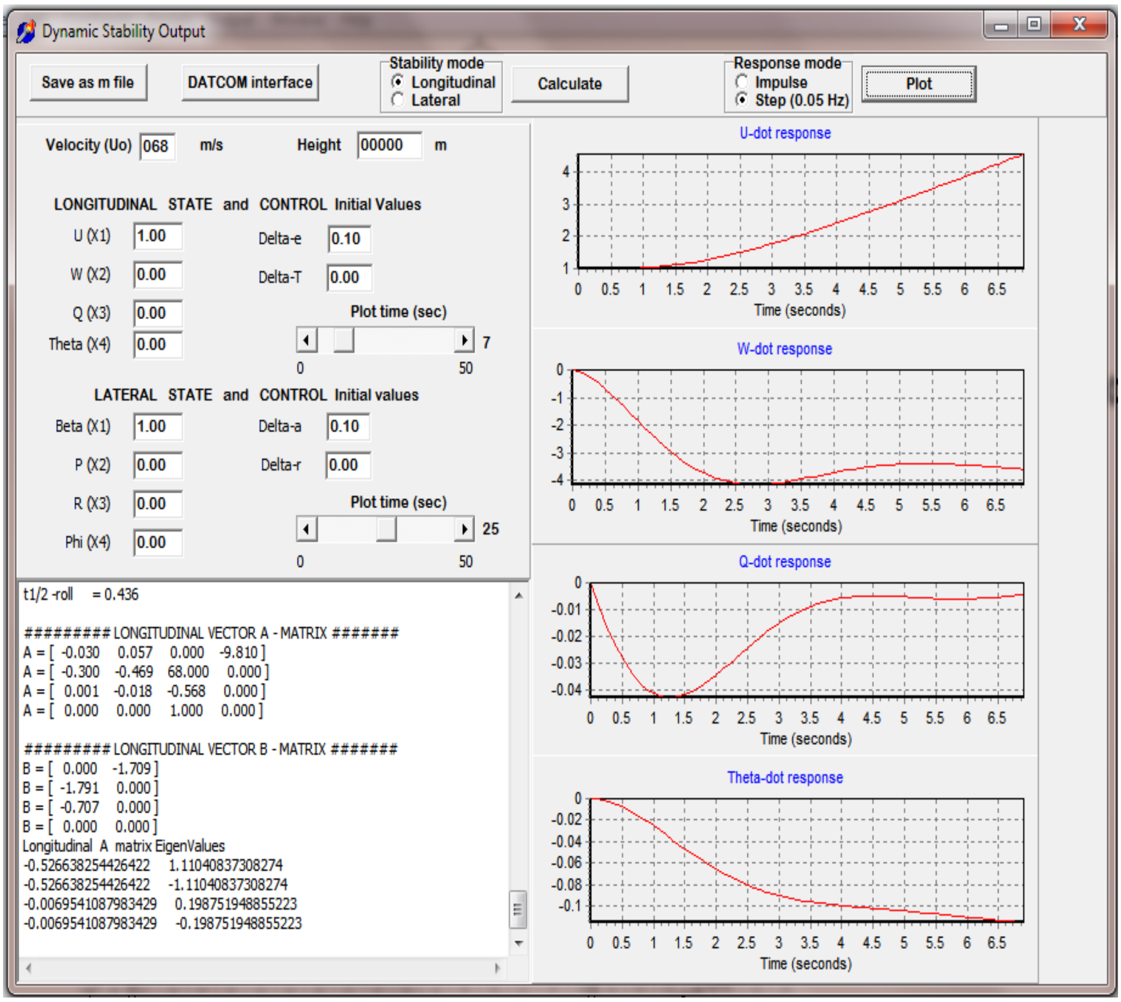

Figure 4-22: Dynamic stability module output form

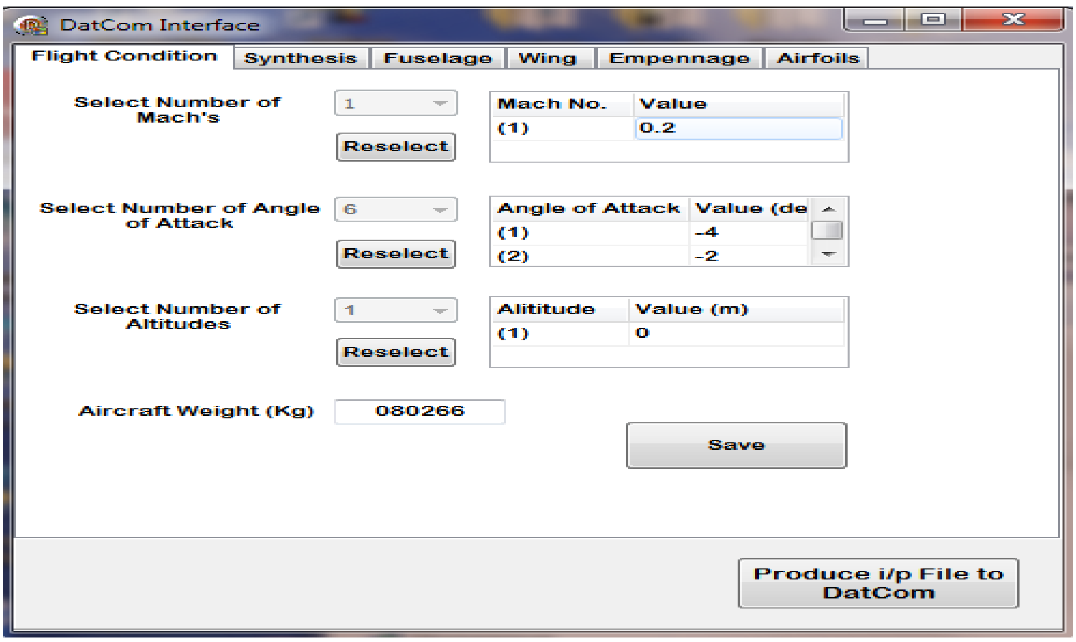

Figure 4-23: DATCOM interface form

4.8 Conclusions

A software package for undergraduate teaching has been developed, to be known as iADS (Interactive Aircraft Design Software). Torenbeek's aircraft design formulae are used as the basis for aircraft synthesis. Due to the fact that the GUI is as important as the software computations itself, an OOUI is implemented to provide an ergonomic interface to the software functions.

The synthesis program does everything needed in the preliminary design environment. This includes calculations involving aircraft geometry, weights, CG, aerodynamics, aspects of flight performance, and cost estimation. The output data are displayed in text form, as well as it can be saved as a spreadsheet for further analysis. A parametric studies module is implemented to understand design parameters' variations. The RAE optimiser is also added that allows an optimum design to be arrived at subject to a number of constraints. An additional module is employed to synthesise the takeoff stage in detail. It is essential for students to evaluate both static and dynamic stability behaviour of the aircraft early on in the preliminary design phase.

The dynamic stability module is implemented based on the analytical approach of aircraft equations of motion. This early evaluation, before the detailed design phase takes place, makes the design process more efficient and helps to speed up the design process itself. 3-View of the proposed aircraft is available as well, and changes dynamically with the design variable variations to explore the influence of these variables on the design and geometric configuration of the aircraft.

Estimation of aircraft weight and cost is fundamentally important. It has been shown by many researchers that the cost of the aircraft is dependent on its weight. Therefore, accurate weight estimation in the preliminary aircraft design phase will yield better indication of key economic factors that govern the commercial suitability and viability of the proposed aircraft design, such as seat mile cost and direct operating cost. In the next chapter a new method for estimating the MTOW is presented. It will be shown that the presented method yields a superior MTOW estimation, than methods that have been presented by various researchers in the past.

Chapter FIVE

Alternative Weight and Cost Estimation Modules

5.1 Introduction

Amongst all the design variables used in aircraft design, weight is the most important one. It is the first variable that must be estimated as accurately as possible, not only in its estimation, but through the whole design process. Performance of the aircraft is dependent on the aircraft having a suitable weight in order for it to carry out its intended mission. Cost of aircraft which is another major parameter for customers (airliners) depends mainly on aircraft weight. Therefore, manufacturers try to make the aircraft as light as possible. Accurate weight estimation at an early stage of aircraft design process is a hard and a difficult task. When the detailed design drawings are complete, the aircraft weight can be calculated accurately by evaluating each part and adding up all component weights. The methodologies used for weight estimation are expanded synchronously with the design phases. In the conceptual design phase, these methodologies are very simple in nature and have significant uncertainty [118] which estimate the aircraft weight as maximum takeoff weight (MTOW). In the preliminary design phase, MTOW is broken down into components and sub-components, the methodologies become more complicated and accurate. More specifically, as information becomes more accessible in this phase, the accuracy of prediction is increased from 10-15% to 5-10%. Torenbeek's formulae achieved a good accuracy of about 6% and almost over-weight predictions. Now, the question is how to improve the accuracy to better than 4%? This requires a development of a new methodology. In fact, Roskam [12] describes three different methodologies that yield different values which differ by as much as 25%. What makes the process difficult is the non-availability of data that could be used to compare aircraft component weights. Although the overall weight figures (such as operating empty weight (OEW) and MTOW) are available, there is a scarcity of information on the detailed component, sub-system and system level.

Hence, instead of applying a complete formulae set for one methodology, a modified weight module is suggested as a new approach for accurate weight estimation in the preliminary design phase [119]. This module evaluates each aircraft component weight by applying many formulae of different methodologies and trying at the same time to avoid using any formula that has secondary (additional) variables which may not be available in the early stages of aircraft design. The one that gives the lowest average value is selected to overcome the over-weight estimations of Torenbeek's approach.

5.2 Modified weight module

Since the body of the aircraft (Wing, Fuselage, and tail) forms 50-60% of the empty weight, in methodology three different formulae sets that define component weights are used and the one that gives the lowest average value is selected. The main input variables that are used in this module are: $m_{to}, S_w, V_{dive,}\ l_{fus}, D_{fus}$, and A_w. Other input variables such as: $N_{ult},\ b_w$, and $S_{w_{fus}}$, are functions of the foresaid variables. If composite or other advanced materials are used, then an allowance is made by applying suitable user-controlled factors to each individual weight component. These factors are used to overcome the shortcomings of some empirical methodologies as mentioned above. For the reason that all formulae work in terms of mass rather than weight, some traditional weight-style abbreviations such as: OEW and MTOW are used interchangeably for convenience. SI units are used unless otherwise mentioned. In order to calculate component weights, pre-calculations for the load factors (limit and ultimate) are required as indicated in chapter three.

The weight module evaluates the aircraft weight (MTOW) by breaking it down into:

1. **Fuel weight**. This is an input design variable.
2. **Zero fuel weight**. This consists of **payload** and **operating empty** weights.

a. **Payload**. This is calculated according to FAA regulations which suggest that passenger weights include *169 lbs* per passenger plus *10 lbs* for winter clothing and *16 lbs* of carry-on bags and personal items for a total of *195 lbs* per passenger. An additional *30 lbs* is assumed for checked bags, leading to the total of *225 lbs* per passenger. This is higher than what has been assumed in the past and based on recent surveys of passenger weights. The aircraft may also carry cargo as desired. An added cargo weight of 40 lbs per passenger is reasonable in the determination of maximum zero fuel weight. Therefore, the total weight per passenger is *265 lbs* or *120 kg*:

$$m_{pay} = 120 \times N_{pas} \tag{5.1}$$

b. **Operating empty weight.** This consists of **operating items, flight crew, flight attendants,** and **empty** weight.

1. **Operating items**. Torenbeek's formula [10] for a short range aircraft is:

$$m_{op\_it} = 8.617 \times N_{pas} \tag{5.2}$$

While for long range aircraft, the formula is:

$$m_{op\_it} = 14.97 \times N_{pas} \tag{5.3}$$

2. **Flight crew.** Torenbeek [10] suggests an average 93 kg per flight crew, i.e.:

$$m_{fl_{crew}} = 93 \times N_{fl_{crew}} \tag{5.4}$$

3. **Flight attendants.** Typically, there are 30 passengers per attendant and Torenbeek [10] suggests 68 kg per flight attendant:

$$m_{fl_{att}} = 68 \times N_{fl_{att}} \tag{5.5}$$

4. **Empty weight.** This weight consists of **wing, fuselage, tail, propulsion, landing gear, surface control, systems,** and **furnishings** components:

a. **Wing:** Wing weight represents about 17-27% of the empty weight. The following formulae (5.6, 5.7, & 5.8) are from Kroo [120], Torenbeek [10], and Raymer [14] respectively:

$$m_w = 4.22 \times S_w + 1.642 \times 10^{-6} \times \frac{N_{ult} \times {b_w}^3 (1+2\times\lambda_w)\times\sqrt{m_{to}\times m_{zf}}}{t_{c_w} \times cos^2\Lambda \times S_w\,(1+\lambda_w)} \tag{5.6}$$

$$m_w = 0.00667 \times N_{ult}^{0.55} \left(t_{c_w} \times c_{r_w}\right)^{-0.3} \times \left(\frac{b_w}{cos\Lambda_{\frac{1}{2}}}\right)^{1.05} \times \left(1 + \sqrt{\frac{1.905\times cos\Lambda}{b_w}}\right) \times \left(\frac{m_{zf}}{S_w}\right)^{-0.3} \times m_{zf} \tag{5.7}$$

$$m_w = 0.0051 \times {S_w}^{0.649} \times {S_f}^{0.1} \times \frac{(N_{ult}\times m_{to})^{0.557}\,(1+\lambda_w)^{0.1}\,{A_w}^{0.5}}{{t_{c_w}}^{0.4} \times cos\Lambda} \tag{5.8}$$

Note that formulae (5.6 & 5.8) are in Imperial units.

These formulae are applied for the current aircraft and Raymer's formula is selected for the reason that it gives the lowest average value as shown in **Figure 5-1**.

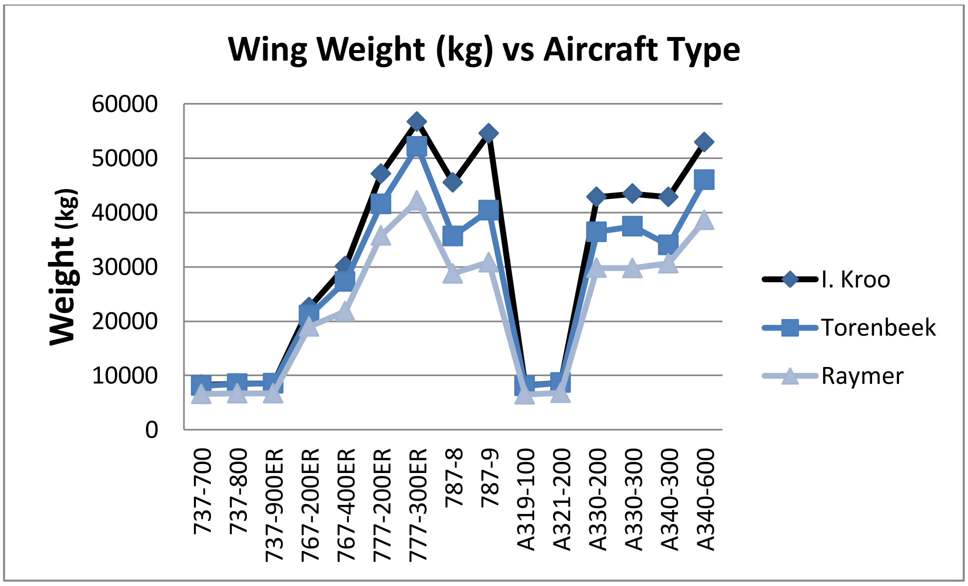

Figure 5-1: Wing weight from three different formulae

b. **Fuselage:** The formulae (5.9, 5.10, & 5.11, respectively) of Nicolai [121], Torenbeek [10], and Raymer [14] are used to calculate the fuselage weight as follows:

$$m_{fus} = 0.0737 \times \left(2 \times D_{fus} \times V_{dive}^{0.338} \times l_{fus}^{0.857} \times (m_{to} \times N_{ult})^{0.286}\right)^{1.1} \tag{5.9}$$

$$m_{fus} = 0.23 \times S_{w_{fus}}^{1.2} \times \sqrt{\frac{V_{dive} \times l_{fus}}{2 \times D_{fus}}} \tag{5.10}$$

$$m_{fus} = 0.4886 \times \sqrt{m_{to} \times N_{ult}} \times l_{fus}^{0.25} \times S_{w_{fus}}^{0.302} \times (1 + k_{ws})^{0.4} \tag{5.11}$$

Where: $$k_{ws} = 0.75 \times \frac{\left(\frac{(1+2\times\lambda_w)}{(1+\lambda_w)}\right) \times (A_w \times S_w)^2 \times tan\Lambda}{l_{fus}} \tag{5.12}$$

Note that formula (5.11) is in Imperial units.

Typically, Raymer’s formula gives the lowest average value as in **Figure 5-2**.

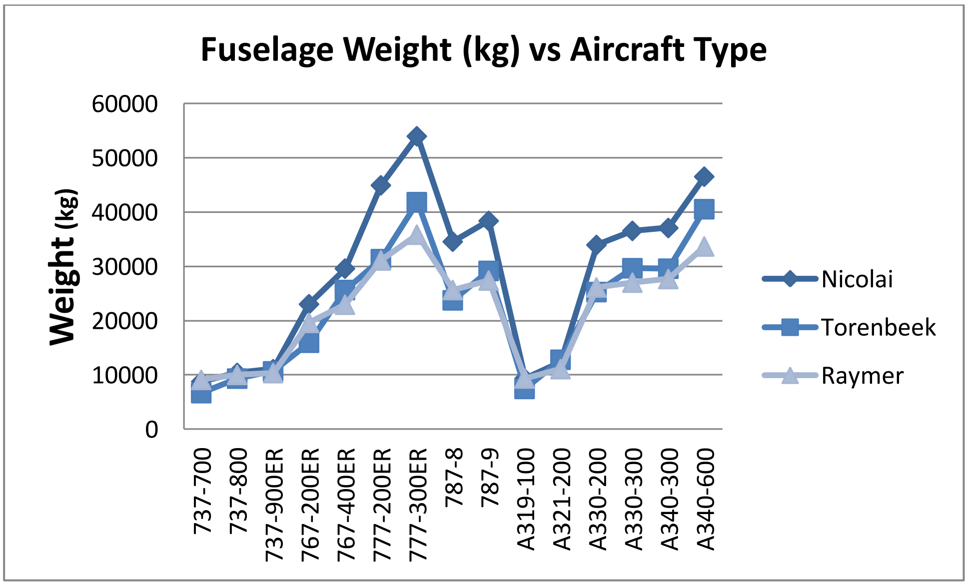

Figure 5-2: Fuselage weight from three different formulae

c. **Tail:** Similar to the wing weight estimation, the formulae of Kroo [120], Torenbeek [10], and Raymer [14] (5.13, 5.14, & 5.15 respectively) are used to calculate both the horizontal and vertical tail weights as in the following:

$$m_{ht} = 5.25 \times S_{x_{ht}} + 0.8 \times 10^{-6} \times \frac{N_{ult} \times b_{ht}^{3} \times m_{to} \times \bar{c} \times \sqrt{S_{x_{ht}}}}{t_{c_{ht}} \times cos\Lambda^{2} \times l_{ht} \times S_{ht}^{1.5}} \tag{5.13a}$$

$$m_{vt} = 2.62 \times S_{vt} + 1.5 \times 10^{-5} \times \frac{N_{ult} \times b_{vt}^{3} \times \left(8.0 + 0.44 \times \frac{m_{to}}{S_{w}}\right)}{t_{c_{vt}} \times cos\Lambda^{2}} \tag{5.13b}$$

$$m_{T} = m_{ht} + m_{vt} \tag{5.13}$$

$$m_{T} = 0.051 \times \frac{V_{dive} \times (S_{ht} + S_{vt})^{1.2}}{\sqrt{cos\Lambda}} \tag{5.14}$$

$$m_{ht} = 0.0379 \times m_{to}^{0.639} \times N_{ult}^{0.1} \times l_{ht}^{-1.0} \times S_{ht}^{0.75} \times (0.3 \times l_{ht})^{0.704} \times cos\Lambda^{-1.0} \times A_{ht}^{0.166} \times \left(1 + \frac{D_{fus}}{b_{ht}}\right)^{-0.25} \times \left(1 + \frac{S_{e}}{S_{ht}}\right)^{0.1} \tag{5.15a}$$

$$m_{VT} = 0.0026 \times m_{to}^{0.556} \times N_{ult}^{0.1} \times l_{vt}^{-0.5} \times S_{vt}^{0.5} \times (0.3 \times l_{vt})^{0.875} \times A_{vt}^{0.35} \times cos\Lambda_{v}^{-1.0} \times t_{c_{vt}}{}^{-0.5} \times \left(1 + \frac{H_{t}}{H_{v}}\right)^{0.225} \tag{5.15b}$$

$$m_{T} = m_{ht} + m_{vt} \tag{5.15}$$

Note that all formulae are in Imperial units except formula (5.14). Again, Raymer's formula gives the lowest average value as shown in **Figure 5-3**.

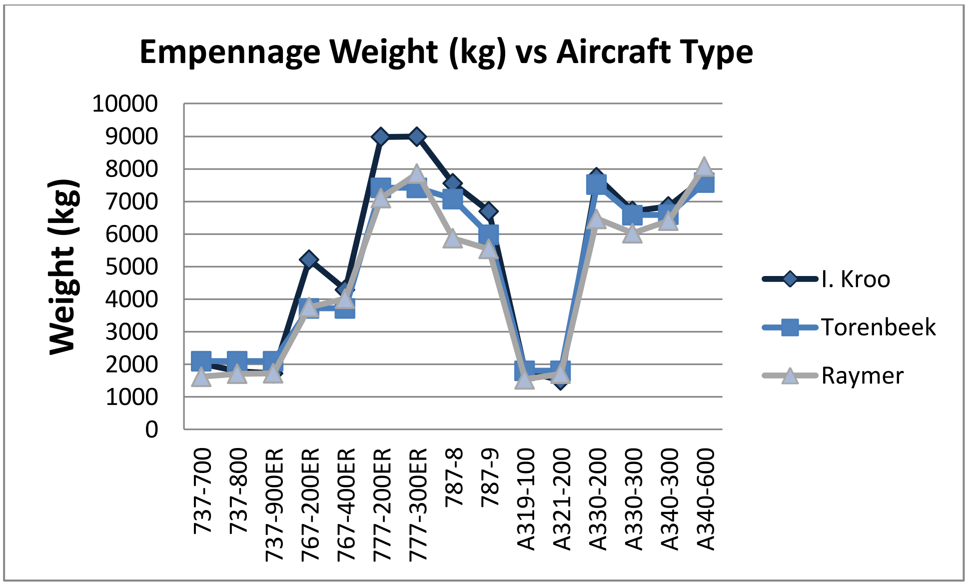

Figure 5-3: Tail Weight from three different formulae

d. **Propulsion:** The major factor in evaluating the weight of the propulsion group (propulsion system & nacelles) is the engine dry weight. In the original weight estimation module, engine dry weight was an input design variable. An alternative way to estimate this weight accurately is done by using the engine data given by Harris [122] and after curve fitting the data, the following two equations were determined:

$$m_{eng} = 0.4054 \times {T_{eng}}^{0.9255} \qquad for\ T_{eng} < 10000\ lbs \tag{5.16}$$

$$m_{eng} = 0.616 \times {T_{eng}}^{0.886} \qquad for\ T_{eng} > 10000\ lbs \tag{5.17}$$

The weight of the propulsion system includes the engines, exhaust, thrust reverser, starting, controls, lubrication, and fuel systems. Torenbeek [10] suggests the following formula for estimating this weight:

$$m_{pro\_sys} = 1.377 \times m_{eng} \times N_{eng} \tag{5.18}$$

While his formula for nacelle group weight is:

$$m_{nac} = 0.055 \times N_{eng} \times N_{eng} \tag{5.19}$$

The total weight of propulsion group is:

$$m_{pro} = m_{pro\_sys} + m_{nac} \tag{5.20}$$

Note that all weights in this sub-section are in pounds (*lbs*).

e. **Landing gear:** The total landing gear weight which includes structure, actuating system, and rolling assembly, is about 3.5-4% of MTOW for aircraft whose weight exceeds 4500 kg [14]. Landing gear weight estimation can be broken down into main gear weight and nose gear weight. The following formulae developed by Torenbeek [10] are employed due to their good estimation (around 3.7% of MTOW):

$$m_{mg} = 40 + 0.16 \times m_{to}^{0.75} + 0.019 \times m_{to} + 1.5 \times 10^{-5} \times m_{to}^{1.5} \tag{5.21}$$

$$m_{ng} = 20 + 0.1 \times m_{to}^{0.75} + 2 \times 10^{1.5} \times m_{to}^{1.5} \tag{5.22}$$

The total weight is:

$$m_{uc} = m_{mg} + m_{ng} \tag{5.23}$$

Note that all weights are in Imperial units.

f. **Surface controls:** The weight of the surface controls comprises the systems associated with control surface actuation and depends mainly on the tail area, Torenbeek [10] suggests the following formula related to takeoff weight instead:

$$m_{sur} = 0.4915 \times m_{to}^{2/3} \tag{5.24}$$

Add 20% for leading flaps or slots and 15% for control dampers if used.

g. **Systems:** Different analysts have their different categories for system component weights. Therefore, it is better to select only one formula set from any one analyst. Raymer's set [14] for example, is good but requires much detailed information which may not be available at the early design stage. Torenbeek's set [10] has been used for a long time and hence it is used here. However, it is most likely to over-estimate the systems weight for the current generation of aircraft, mainly because of avionics and materials. Systems are broken down into seven sub-categories as follows:

g.1 **Auxiliary power unit (APU):** The installed APU weight is dependent mainly on the dry engine weight of APU as in the following formula:

$$m_{apu} = 2.2 \times m_{apu_{dry}} \tag{5.25}$$

In the absence of the uninstalled APU weight, Kundu's formula [93] is used:

$$m_{apu_{dry}} = 0.001 \times m_{to} \tag{5.26}$$

g.2 **Instruments and Avionics:** This weight is estimated based on both takeoff weight and stage length:

$$m_{ins} = 0.347 \times \left(\frac{m_{to}}{2}\right)^{0.555} \times \left(\frac{s_l}{1000}\right)^{0.25} \tag{5.27}$$

g.3 **Hydraulics and Pneumatics:** The weight of hydraulic systems is related directly with the takeoff weight:

$$m_{hyd} = 0.015 \times \left(\frac{m_{to}}{2}\right) + 272 \tag{5.28}$$

g.4 **Electrical system:** This weight depends only on cabin length ($l_{2_{fus}}$) and fuselage diameter (D_{fus}):

$$m_{ele} = 10.8 \times \left(\pi \times l_{2_{fus}} \times \left(0.9 \times D_{fus}\right)^2\right)^{0.7} \times \left(1 - 0.18 \times \left(\pi \times l_{2_{fus}} \times \left(0.9 \times D_{fus}\right)^2\right)^{0.7}\right) \tag{5.29}$$

Note that this formula is in Imperial units.

g.5 **Air conditioning and Anti-icing:** Again this weight depends on cabin length ($l_{2_{fus}}$) only:

$$m_{acic} = 14 \times l_{l_{fus}}^{1.28} \tag{5.30}$$

g.6 **Oxygen system:** This weight is added to the modified module which relates to cruise altitude and range. If the altitude is less than 25000 feet, the following formula is used:

$$m_{oxy} = 20 + 0.5 \times N_{pas} \tag{5.31}$$

If the altitude is higher than 25000 feet, the following formulae are used:

$$m_{oxy} = 30 + 1.2 \times N_{pas} \qquad for\ short\ range \tag{5.32}$$

$$m_{oxy} = 40 + 2.4 \times N_{pas} \qquad for\ long\ range \tag{5.33}$$

g.7 **Paint and Miscellaneous:** Other weight is added to the modified module which represents 0.006 of the take-off weight:

$$m_{pnt} = 0.006 \times m_{to} \tag{5.34}$$

Therefore, the **total systems weight** is:

$$m_{sys} = m_{apu} + m_{ins} + m_{hyd} + m_{ele} + m_{acic} + m_{oxy} + m_{pnt} \tag{5.35}$$

f. **Furnishings:** Furnishings are mainly proportional to the number of actual passenger seats. For a more accurate calculation, this weight is based on the actual division of seats between first class and economy class. In the early stages of the aircraft design process, the maximum number of seats of one class is used. Torenbeek's formula [10] for estimating this depends on zero fuel weight, and is given as:

$$m_{furn} = 0.196 \times m_{zf}^{0.91} \tag{5.36}$$

Hence, **Empty Weight** is the sum of all structural component weights. i.e.:-

$$m_e = m_w + m_{fus} + m_T + m_{uc} + m_{pro} + m_{sur} + m_{sys} + m_{furn} \tag{5.37}$$

5.2.1 Case study

In the preliminary design phase, improving the weight estimate is a difficult process due to the non-availability of data that could be used in the estimation of maximum takeoff weight. Although overall weight figures are available, there is a scarcity of information regarding detailed component, sub-system and system level weights. Operating empty weight (m_{oe}) and maximum takeoff weight (m_{to}) are the only available data for the existing aircraft. These data are used in a case study to assess the modified module. **Table 5-1** shows the output results of the Torenbeek's weight estimation method described in chapter 3, whilst **Table 5-2** shows the results for the modified weight estimation module described above.

On examination of **Table 5-1**, we can conclude that the absolute average difference in operating empty weight is 5.14%, and 3.12% for maximum takeoff weight. At first instance these figures seem very good for the preliminary design phase. More specifically, as aircraft weight is reduced, these differences are reduced as well. This explains why Torenbeek's formulae have been in use for a long time, mainly for its simplicity and accuracy of prediction.

Aircraft Type	Published Data		Calculated Data		% Diff. m_{oe}	% Diff. m_{to}
	m_{oe}	m_{to}	m_{oe}	m_{to}		
A319 – 100	40800	75500	40773	76525	- 0.06	+ 1.36
A321 - 200	48500	95510	52487	99932	+ 8.22	+ 4.63
A330 – 200	119600	238000	125006	240683	+ 4.52	+ 1.13
A330 – 300	124500	235000	131967	246856	+ 6.00	+ 5.05
A340 – 300	130200	276500	131979	283368	+ 1.37	+ 2.48
A340 – 600	177800	368000	174859	363364	- 1.68	- 1.28
737 – 700	38147	70305	38643	72053	+ 1.3	+ 2.49
737 – 800	41145	79245	43974	83192	+ 6.88	+ 4.98
737 – 900ER	44676	85130	47075	88918	+ 5.37	+ 4.45
767 – 200ER	84280	179625	90966	185825	+ 7.93	+ 3.45
767 – 400ER	103145	204570	111749	211826	+ 8.34	+ 3.55
777 – 200ER	145015	297550	151545	302435	+ 4.5	+ 1.64
777 – 300ER	167830	351500	185722	365633	+ 10.66	+ 4.02

Table 5-1: Torenbeek weight estimation method

Aircraft Type	Published Data		Calculated Data		% Diff. m_{oe}	% Diff. m_{to}
	m_{oe}	m_{to}	m_{oe}	m_{to}		
A319 – 100	40800	75500	38918	74670	- 4.63	- 1.10
A320 - 200	42600	78000	43909	81046	+ 3.07	+ 3.91
A321 - 200	48500	93510	46934	94879	- 3.23	+ 1.46
A330 – 200	119600	238000	117101	232778	- 2.09	- 2.19
A330 – 300	124500	235000	118746	233636	- 4.62	- 0.58
A340 – 300	130200	276500	124116	275505	- 4.67	- 0.36
A380 – 800	276800	571000	264111	571645	- 4.58	+ 0.11
737 – 700	38147	70305	36664	70074	- 3.89	- 0.33
737 – 800	41145	79245	41294	80512	+ 0.36	+ 1.6
737 – 900ER	44676	85130	43277	85121	- 3.13	- 0.01
767 – 200ER	84280	179625	86626	181484	+ 2.78	+ 1.03
767 – 400ER	103145	204570	99113	199189	- 3.91	- 2.63
777 – 200ER	145015	297550	139771	290660	- 3.62	- 2.32
777 – 300ER	167830	351500	164944	345056	- 1.72	- 1.83

Table 5-2: Modified weight estimation module

From **Table 5-2** it can be seen that the absolute average difference in operating empty weight (m_{oe}) is 3.45%, and 1.06% for maximum takeoff weight (m_{to}). These figures show an improvement over the original methodology. The modified module improves weight estimation by 33% for operating empty weight (m_{oe}) and 66% for maximum takeoff weight (m_{to}) in terms of absolute average difference. The comparison of the predicted MTOW's, with actual published MTOW's of old and new aircraft confirms that the proposed methodology works very well, with predictions of the MTOW better than 3% in most cases which is a vast improvement on the existing MTOW estimation methods in the preliminary design phase.

5.3 Cost estimation module

Direct operating cost and seat mile cost are significant parameters in evaluating competitive aircraft designs. Although the rule of thumb is that the aircraft cost depends mainly on aircraft empty weight, occasionally this is not right. For example, using new technologies and materials (e.g. composite) makes the aircraft lighter, but the production cost is increased. Therefore, manufacturers always pick the design and price that maximises their own return. This requires better estimates of the operating costs (OC) and a good measure of the market elasticity. Customers are interested in cost savings, not just low aircraft price at the time of purchase but also throughout the lifetime of the aircraft [123]. More specifically, one pays for a pound of aluminium in the wing once, but a pound of fuel on every flight [124].

Aircraft operating costs consist of many contributing costs which are due to depreciation, insurance, maintenance, fuel burn, flight crew, cabin crew, landing fees, and passenger services. These items are grouped into two main categories which are direct operating cost (DOC) and indirect operating cost (IOC). IOC is difficult to estimate well, since it depends on the services that the airline (customer) offers [124]. Therefore, DOC is a very useful and widely-used parameter for comparative analysis.

In 1944, the Air Transportation Association of America (ATA) developed the first set of equations to estimate DOC. It continued periodically to revise these formulae to match current statistical cost data. The last updated version was published in 1967 [97]. Many methodologies have been developed thereafter [125] [126]. The purpose of applying a standard methodology to estimate DOC is to enable both the manufacturer and the customer to assess the economic suitability of the aircraft operation on a given route. ATA pointed out that it "*must essentially be general in scope, and for simplicity should preferably employ standard formulae into which the values appropriate to the aircraft under study are substituted*" [97]. Typically, aircraft manufacturers use standard methodologies in their cost comparisons, while customers (airlines) always generate their own methodologies based on many things that may not be accounted for, such as fleet size, route structure, accounting procedures, etc, or capitalise certain costs which then can be reported in depreciation or amortisation cost figures.

5.4 Extended cost estimation module

A cost module based on the AEA methodology was incorporated into the software as described in the previous chapter. It evaluated the aircraft price based on statistical data, DOC per flight, and seat-mile cost. Instead of implementing one standard methodology (AEA [98]) to analyse DOC, two further common methodologies (ATA [97] and NASA [127]) have been incorporated into this cost module. Their formulae are explained in detail in the DOC components section (5.4.1). ATA, the professional society of airliner business in the U.S., used industry-wide statistical data to develop a standard methodology for estimating comparative DOC for jet aircraft. NASA's methodology is an estimation known as DOC+I (Direct Operating Cost plus Interest). It is based on the work done by Liebeck [127], who was able to draw upon the operating costs of McDonnell Douglas aircraft in commercial service up until 1993. It is therefore based on a more recent set of data which reflect airline costs in a deregulated environment. AEA methodology has been accepted as the basis for comparison in Europe. These methodologies depend initially on estimating aircraft price (capital cost).

Estimating aircraft price in the early stages of aircraft design requires an investigation of the actual data available. Prices of the current Boeing [128] and Airbus [129] aircraft at year 2010 are shown in **Table 5-3** and **Table 5-4**, respectively. These prices are plotted as a function of their empty weights in **Figure 5-4**. Prices are shown as an average, since the exact price of a given aircraft depends upon special equipment particular to different buyers.

Airplane Families	2010 $ in Millions Average
737 Family	
737-600	56.9
737-700	67.9
737-800	80.8
737-900ER	85.8
747 Family	
747-8	317.5
747-8 Freighter	319.3
767 Family	
767-200ER	144.1
767-300ER	164.3
767-300 Freighter	167.7
767-400ER	180.6
777 Family	
777-200ER	232.3
777-200LR	262.4
777-300ER	284.1
777 Freighter	269.1
787 Family	
787-8	185.2
787-9	218.1

Table 5-3: Aircraft prices for various Boeing aircraft in 2010, **[128]**

Airplane Type	2010 $ in Millions Average
A318	63.2
A319	74.7
A320	82.0
A321	95.7
A319/A320/A321: new engine option average	6.2
A330-200	193.8
A330-200 Freighter	196.6
A330-300	215.5
A340-300	231.0
A340-500	253.8
A340-600	267.4
A350-800	228.6
A350-900	259.6
A350-1000	290.7
A380-800	365.3

Table 5-4: Aircraft prices for various Airbus aircraft in 2010, **[129]**

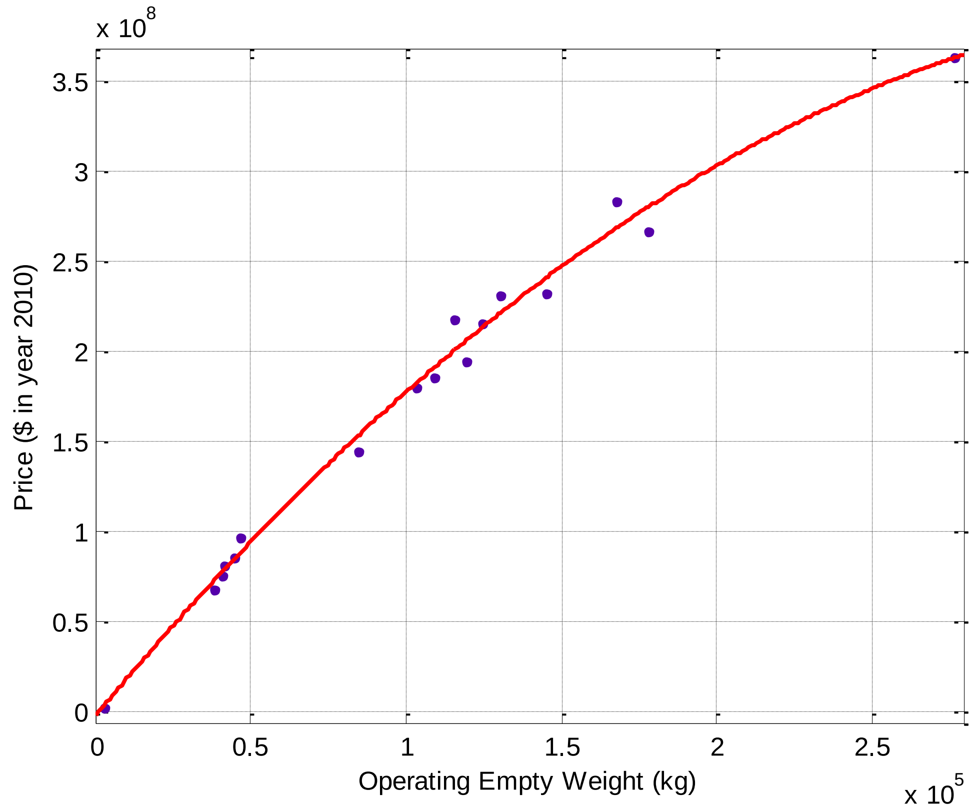

Figure 5-4: Airbus & Boeing aircraft prices vs. Operating empty weights

Although, Sforza [130] was interested in the specific cost of the aircraft in terms of \$/lb, it is easier to evaluate the aircraft price (in \$) using the operating empty weight (m_{oe}) directly as in the following:

$$C_{AC} = 10^6 \times (1.18 \times m_{oe}^{0.48} - 116) \quad , \; if \; m_{oe} \geq 10000 \; kg \tag{5.38}$$

$$C_{AC} = -0.002695 \times {m_{oe}}^2 + 1967 \times m_{oe} - 2158000 \quad , \; if \; m_{oe} < 10000 \; kg \tag{5.39}$$

A similar procedure has been adopted by Kroo [120] to estimate the price of the aircraft's engine as shown in **Figure 5-5**. These prices are based on data available and applicable in 1990 and can be corrected to figures pertinent to 2010, by applying a simple inflation multiplier (1.76) which is the ratio of the consumer price index (CPI) for year 2010 to that for 1990. Reference [131] presents some deflators that are used in the aerospace industry while more extensive information on the CPI and other economic factors may be found in [132]. The formula for engine price (in $) is:

$$C_{eng} = 1.76 \times 82.5 \times T_{eng} \tag{5.40}$$

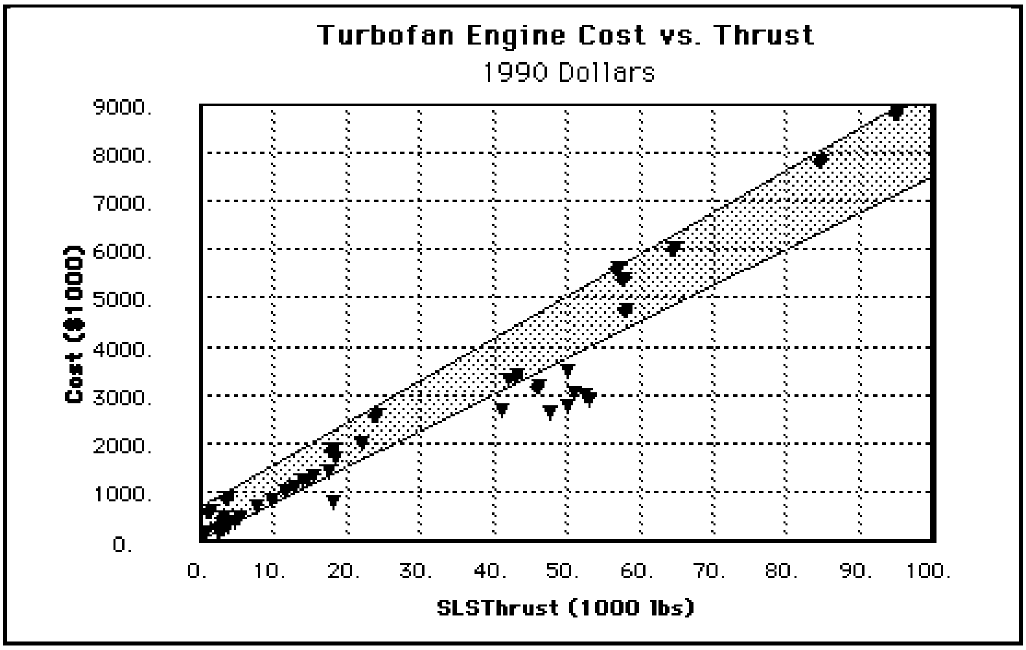

Figure 5-5: Engine prices vs. SLS thrust (lbs), [120]

5.4.1 DOC components

DOC is expressed in terms of $/hour, $/mile, ¢/seat-mile, or for cargo aircraft, ¢/ton-mile. Costs in terms of $/mile indicate the maximum loss of the airliner with an empty aircraft, while costs per unit productivity such as ¢/seat-mile, or ¢/ton-mile indicate the fare that must be charged with reasonable load factors. DOC is broken down into its components and is explained in the following sub-sections. Each component cost is computed using the three methodologies: ATA, NASA, and AEA. Note that all component costs are per trip and some of them are based on the evaluation of the annual utilisation (U) of the aircraft, which in turn depends mainly on the customer and its route (i.e. the range). The latter can be derived in terms of block hour time (t_B). The original ATA graph [97], shown in **Figure 5-6**, is represented by the following formula:

$$U = 6100 - 3100 \times t_B^{-0.3342} \tag{5.41}$$

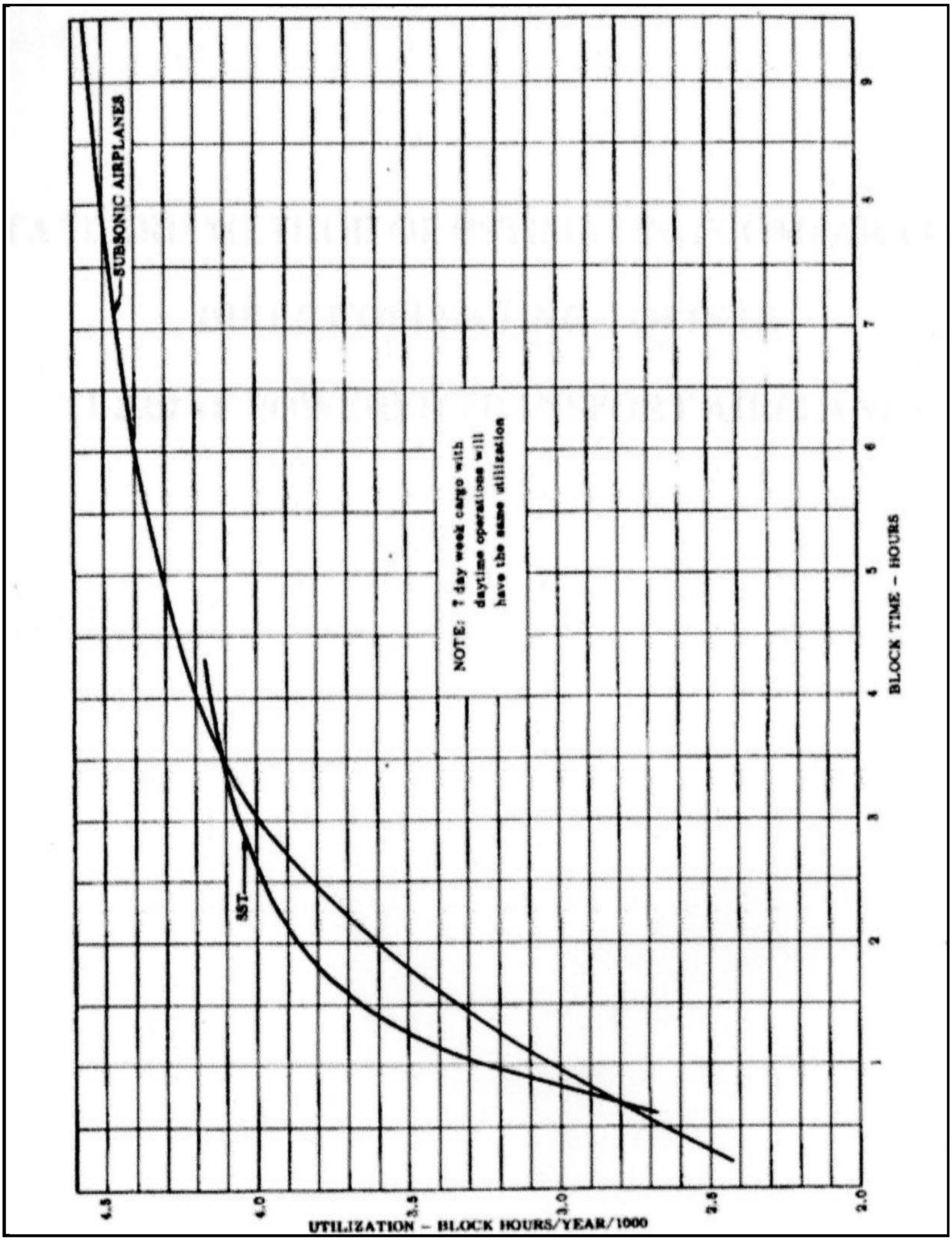

Figure 5-6: Original ATA graph for utilisation, [97]

MIT [133] collated the daily utilisation for a number of US airliners in year 2006 as shown in **Figure 5-7**. The average utilisation from **Figure 5-7** is determined to be, 10.64 hours/day and in turns the annual utilisation is about 3800 hours/year. It seems approximately equal to the average of the original ATA graph.

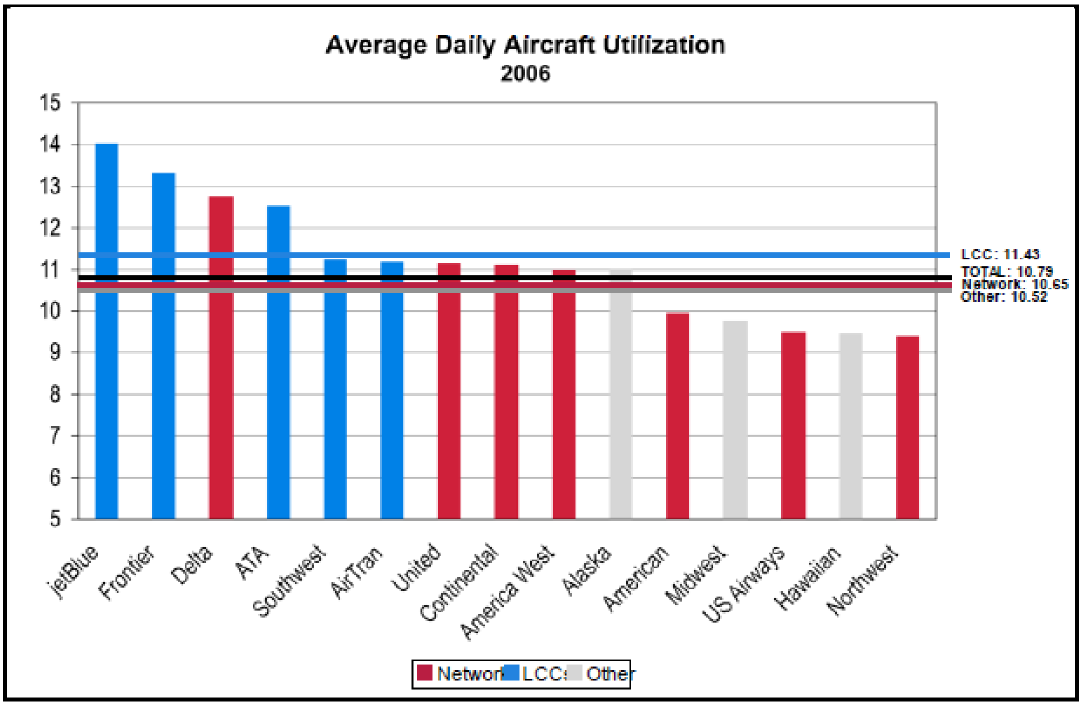

Figure 5-7: Daily utilisation in the USA for year 2006, [133]

NASA suggests values of utilisation as trips per year as:

Short range aircraft = 2100 trips/year

Medium range aircraft = 625 trips/year

Long range aircraft = 480 trips/year

For short and medium ranges, AEA utilisation (U) formula in terms of hours/year is:

$$U = \left(\frac{3750}{t_B + 0.5}\right) \times t_B \quad (5.42)$$

While for long range, it is assumed to be equal to 4800 hours/year.

5.4.1.1 Depreciation

The Depreciation of the capital value of an aircraft is dependent to a large degree on the individual airline and its competitive conditions as the aircraft is maintained in a fully airworthy condition throughout its life. ATA depreciation period (D_a) is 12 years and 0% is the residual value for subsonic aircraft and its components [97]. The ATA depreciation formula is:

$$C_{dp} = \frac{(C_{AC}+0.1\times C_{AF}+0.4\times C_{eng})\times t_B}{D_a\times U} \tag{5.43}$$

NASA's formula for determining depreciation is:

$$C_{dp} = \frac{C_{(dp)_{year}}}{U} \tag{5.44}$$

Where $C_{(dp)_{year}}$ is evaluated using the following formula:

$$C_{(dp)_{year}} = (1-R)\times\left(\frac{C_{AF}}{P_{af}}\right)+S_{af}\times\left(\frac{C_{AF}}{P_{af}}\right)+\frac{C_{eng}}{P_{af}}+S_{eng}\times\left(\frac{C_{eng}}{P_{eng}}\right) \tag{5.45}$$

AEA suggests a ten-aircraft fleet with a 14-year lifespan and a residual value (R) of 10% of the total investment. i.e.:

$$C_{dp} = \frac{0.9\times t_B\times(C_{AC}+0.1\times C_{AF}+0.3\times C_{eng})}{14\times U} \tag{5.46}$$

5.4.1.2 Hull Insurance

ATA insurance value per trip [97] is determined as follows:

$$C_{ins} = \frac{t_B\times R_{ins}\times C_{AC}}{U} \tag{5.47}$$

Where: $R_{ins} = 0.0023$, as in **Figure 5-8** [134].

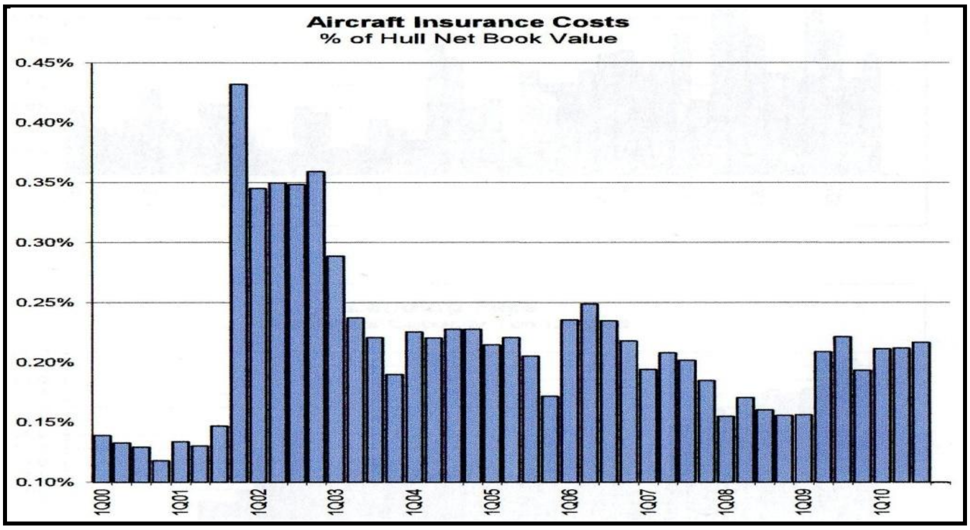

Figure 5-8: ATA insurance rate for years 2000- 2010, [134]

NASA's formula [127] for determining the hull insurance is:

$$C_{ins} = \frac{0.0035 \times C_{AC}}{U} \tag{5.48}$$

AEA's formula for determining the hull insurance is:

$$C_{ins} = \frac{0.005 \times C_{AC}}{U} \tag{5.49}$$

5.4.1.3 Interest

Although the original ATA method did not include the interest cost, most aircraft purchases nowadays are financed through the use of long-term debt and a down payment from company funds. For this reason, Hays [135], suggests the following AEA formula to be used in ATA methodology with R_{int} typically = 0.07 [134] as in **Figure 5-9**:

$$C_{int} = \frac{t_B \times R_{int} \times (C_{AC} + 0.1 \times C_{AF} + 0.3 \times C_{eng})}{U} \tag{5.50}$$

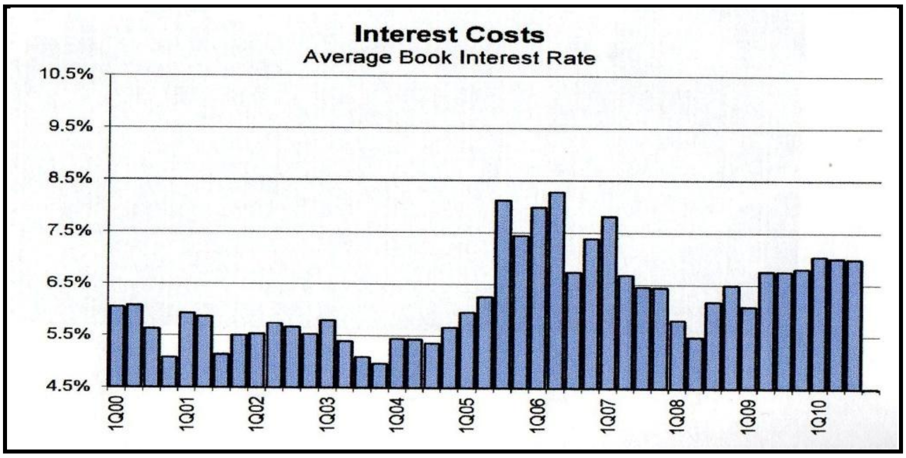

Figure 5-9: ATA interest rate for years 2000- 2010, [134]

The NASA formula for interest cost per trip is:

$$C_{int} = \frac{R_{int} \times (C_{AC} + 0.06 \times C_{AF} + 0.23 \times C_{eng})}{U} \tag{5.51}$$

Where R_{int} = 0.055 [127].

The AEA formula is similar to the ATA formula with R_{int} = 0.053.

5.4.1.4 Flight Crew

ATA's formula is based on economic conditions that prevailed in 1967 and the result has to be updated to 2010 conditions. It is convenient to simply inflate the equation result by the ratio of the consumer price index (CPI) [85] in 2010 to that in 1967 which is:

$$(CPI)_{2010}/(CPI)_{1967} = 218.056/33.4 = 6.53$$

This value can easily be corrected to the $(CPI)_{2013}$. Since the majority of data for costing purposes was available for the period ending 2010, it was decided not to apply the CPI correction for the year 2013.

This factor modifies the ATA's formula to be:

$$C_{fc} = t_B \times \left(\frac{0.326 \times m_{to}}{1000} + 653\right) \tag{5.52}$$

The NASA formula for determining the flight crew cost is:

$$C_{fc} = t_B \times N_{fl_{crew}} \times F_i \times \left(440 + \frac{0.532 \times m_{to}}{1000}\right) \tag{5.53}$$

Where m_{to} is in pounds (*lbs*).

AEA uses $493 per block hour for a two-crew operation, i.e.:

$$C_{fc} = t_B \times 493 \tag{5.54}$$

5.4.1.5 Cabin Crew

ATA cabin crew costs are classified as indirect costs and hence, there is no separate formula for these costs.

NASA's formula for cabin crew cost is:

$$C_{cc} = t_B \times N_{fl_{att}} \times C_{cc_b} \tag{5.55}$$

Where C_{cc_b} = base cabin crew cost of $60/hr for domestic flights and $78/hr for international flights. The AEA formula is similar to NASA except that AEA uses $81/hr for C_{cc_b} .

5.4.1.6 Fuel and Oil

The current fuel prices may be found from the IATA website [136]. A factor of *0.326* is used to convert 1*kg* of fuel weight to 1*gal* of volume for the reason that the density of Jet A fuel may be taken as *6.76 lbs/gal* at standard conditions. On the other hand, an examination of prices for turbine oil shows that it is around *$50/gal*. Therefore, applying simple CPI inflation is sufficiently accurate and the cost of lubricating oils is relatively stable and does not generally follow the rise in fuel prices.

The ATA formula [97] for fuel and oil cost per trip (which includes 2% non revenue flying and assuming that the rate of consumption of oil is *0.135 lbs/hr/engine)*, is:

$$C_{fuel} = 1.02 \times \left(m_{fuel} \times C_{fu} + 0.135 \times t_B \times N_{eng} \times C_{oil}\right) \tag{5.56}$$

Where $C_{fu} = 2.15\ \$/gal$ is the average value for the ten year period, for the first quarter as in **Figure 5-10** [134], and $C_{oil} = 50\ \$/gal$. The figure of *0.135 lbs/hr/engine* i.e., the rate of oil consumption may be changed if a more accurate value is available for the specific engine.

The NASA and AEA formula for determining the cost of fuel is:

$$C_{fuel} = \frac{m_{fuel} \times C_{fu}}{\rho_f} \tag{5.57}$$

Where m_{fuel} is in pounds and excluding reserves, while $\rho_f = 6.7\ lbs/gal$.

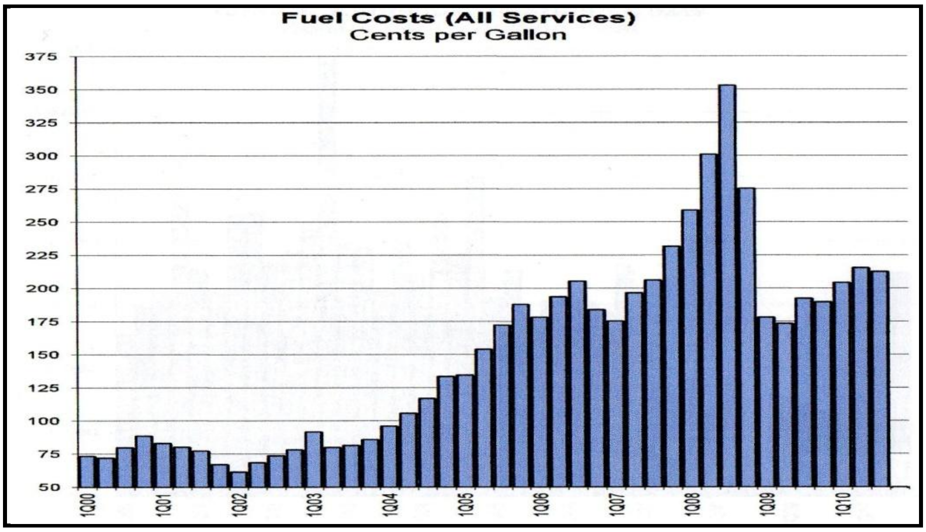

Figure 5-10: First quarter fuel cost for years 2000- 2010, [134]

5.4.1.7 Maintenance

This term includes labour and material costs for both airframe and engines. Furthermore, burden costs are also included i.e.:

$$C_{maint} = (C_{al} + C_{amm}) + (C_{el} + C_{emm}) + C_{bur} \tag{5.58}$$

5.4.1.7.1 Airframe labour cost

The ATA formula for determining the labour cost associated for maintaining the airframe is:

$$C_{al} = \left(C_{(al)_{kh}} \times t_f + C_{(al)_{kc}}\right) \times C_{lr} \times {N_{ma}}^{1/2} \tag{5.59}$$

Where $C_{(al)_{kc}} = \frac{0.05 \times m_{e-e}}{1000} + 6 - \frac{630}{\left(\frac{m_{e-e}}{1000} + 120\right)}$, (5.60)

$$C_{(al)_{kh}} = 0.59 \times C_{(al)_{kc}} \quad , \tag{5.61}$$

$C_{lr} = \$42\ /hr$, as in **Figure 5-11** (assuming average 2000 hours / year), and

$N_{ma} =$ Cruise Mach No.

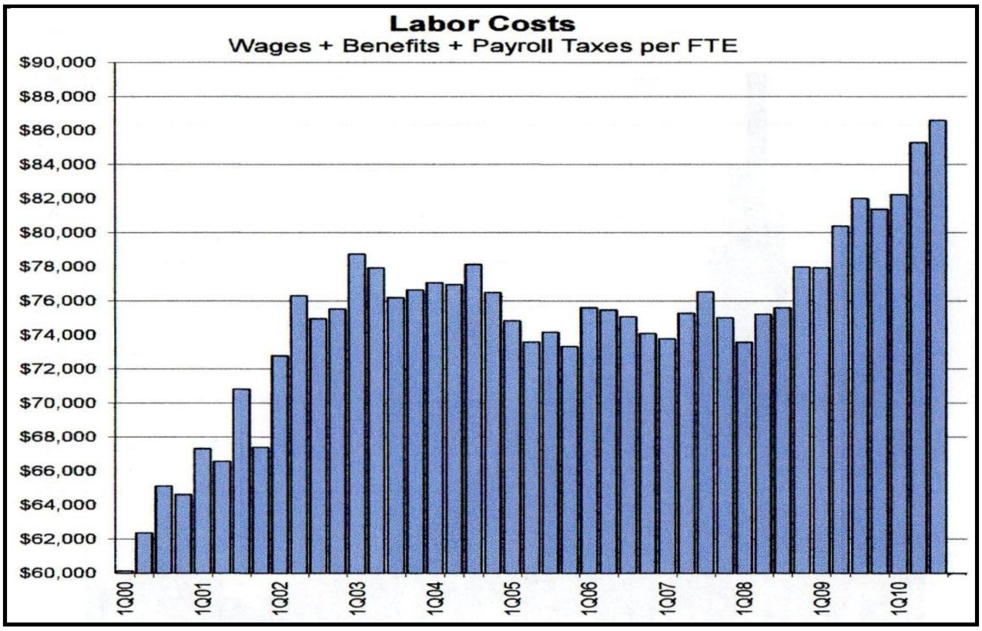

Figure 5-11: Labour cost per person per year for years 2000-2010, [134]

The NASA formula determining the labour cost associated for maintaining the airframe is:

$$C_{al} = \left(\left(1.26 + 1.774 \times \left(\frac{m_{e-e}}{10^5}\right) - 0.1071 \times \left(\frac{m_{e-e}}{10^5}\right)^2\right) \times t_B + \left(1.614 + 0.7227 \times \left(\frac{m_{e-e}}{10^5}\right) + 0.1204 \times \left(\frac{m_{e-e}}{10^5}\right)^2\right)\right) \times C_{lr} \quad (5.62)$$

Where $C_{lr} = 25\,\$/hr$

The AEA formula determining the labour cost associated for maintaining the airframe is:

$$C_{al} = \left(\frac{\left(0.09 \times m_{e-e} + 6.7 - \frac{350}{m_{e-e}+75}\right)\left(0.8 + 0.68 \times (t_B - 0.25)\right)}{t_B}\right) \times C_{lr} \quad (5.63)$$

Where $C_{lr} = 63\,\$/hr$

5.4.1.7.2 Airframe material cost

The ATA formula determining the material cost associated for maintaining the airframe is:

$$C_{amm} = C_{(am)_{kh}} \times t_f + C_{(am)_{kc}} \tag{5.64}$$

Where $C_{(am)_{kh}} = \frac{3.08 \times C_{AF}}{10^6}$, and $C_{(am)_{kc}} = \frac{6.24 \times C_{AF}}{10^6}$

The NASA formula for the material cost associated for maintaining the airframe is:

$$C_{amm} = \left(\left(12.39 + 29.8 \times \left(\frac{m_{e-e}}{10^5}\right) + 0.1806 \times \left(\frac{m_{e-e}}{10^5}\right)^2\right) \times t_B + \left(15.2 + 97.33 \times \left(\frac{m_{e-e}}{10^5}\right) - 2.862 \times \left(\frac{m_{e-e}}{10^5}\right)^2\right)\right) \times 1.509 \tag{5.65}$$

The AEA formula for the material cost associated for maintaining the airframe is:

$$C_{amm} = \left(\frac{4.2 + 2.2 \times (t_B - 0.25)}{t_B}\right) \times \left(\frac{C_{AF}}{10^6}\right) \tag{5.66}$$

5.4.1.7.3 Engines labour cost

The ATA formula for determining the labour cost associated for maintaining the engines is:

$$C_{el} = \left(C_{(el)_{kh}} \times t_f + C_{(el)_{kc}}\right) \times C_{lr} \tag{5.67}$$

Where $C_{(el)_{kh}} = \left(0.6 + \frac{0.027 \times T_{eng}}{10^3}\right) \times N_{eng}$, (5.68)

$$C_{(el)_{kc}} = \left(0.3 + \frac{0.03 \times T_{eng}}{10^3}\right) \times N_{eng} \tag{5.69}$$

The NASA formula for determining the labour cost associated for maintaining the engines is:

$$C_{el} = \left(0.645 + \left(\frac{0.05 \times T_{eng}}{10^4}\right)\left(0.566 + \frac{0.434}{t_B}\right)\right) \times N_{eng} \times C_{lr} \times t_B \tag{5.70}$$

The AEA formula for determining the labour cost associated for maintaining the engines is:

$$C_{el} = 0.21 \times C_1 \times C_3 \times C_{lr} \times \left(1 + T_{eng}\right)^{0.4} \tag{5.71}$$

Where $C_1 = 1.27 - 0.2 \times {E_{bpr}}^{0.2}$, and $C_3 = 0.032 \times N_c + K$

5.4.1.7.4 Engines material cost

The ATA formula for determining the material cost associated for maintaining the engines is:

$$C_{emm} = C_{(em)_{kh}} \times t_f + C_{(em)_{kc}} \tag{5.72}$$

Where $C_{(em)_{kh}} = 2.5 \times N_{eng} \times \left(\frac{C_{eng}}{10^5}\right)$, (5.73)

$$C_{(em)_{kc}} = 2 \times N_{eng} \times \left(\frac{C_{eng}}{10^5}\right) \tag{5.74}$$

The NASA formula determining the material cost associated for maintaining the engines is:

$$C_{emm} = \left(\left(25 + \left(\frac{0.05 \times T_{eng}}{10^4}\right)\right) \times \left(0.62 + \frac{0.38}{t_B}\right)\right) \times N_{eng} \times t_B \times 1.509 \tag{5.75}$$

The AEA formula determining the material cost associated for maintaining the engines is:

$$C_{em} = 2.56 \times C_1 \times (C_2 + C_3) \times \left(1 + T_{eng}\right)^{0.8} \tag{5.76}$$

Where $C_2 = 0.4\left(\frac{E_{oapr}}{20}\right)^{1.3} + 0.4$ (5.77)

$$C_1 = 1.27 - 0.2 \times {E_{bpr}}^{0.2} \tag{5.78}$$

$$C_3 = 0.032 \times N_c + K \tag{5.79}$$

Note that the AEA total engine maintenance (labour + material) formula is:

$$C_{em} = N_{eng} \times (C_{el} + C_{emm}) \times \frac{t_B + 1.3}{t_B - 0.25} \qquad (5.80)$$

5.4.1.7.5 Maintenance burden

This is defined as labour and material overheads that contribute to overall maintenance costs through activities such as administration, controlling, monitoring, planning, testing, and tooling. It is also called "Indirect Maintenance Cost".

The ATA formula for determining the maintenance burden is:

$$C_{bur} = 1.8 \times (C_{al} + C_{el}) \qquad (5.81)$$

The NASA formula for determining the maintenance burden is:

$$C_{bur} = 2 \times (C_{al} + C_{el}) \qquad (5.82)$$

The AEA has no burden cost included.

5.4.1.8 Landing fee

The landing fee is based on the maximum landing weight for domestic operations, or maximum takeoff gross weight for international operations. This may vary significantly in Europe, with possible additional fees such as for NO_x emissions or community noise, which are not included in the DOC. The ATA methodology categorises landing fee as an indirect cost, while the NASA formula is:

$$C_{lf} = 2.2 \times \frac{m_{lan}}{1000} \qquad for\ domestic\ operations \qquad (5.83)$$

$$C_{lf} = 6.25 \times \frac{m_{to}}{1000} \qquad for\ international\ operations \qquad (5.84)$$

Note that the weights (m_{lan}, m_{to}) are in pounds (*lbs*).

The landing fees as determined by the AEA formula is:

$$C_{lf} = 7.8 \times \frac{m_{to}}{1000} \tag{5.85}$$

5.4.1.9 Navigation fee

The navigation fee is based on the first 500nm of a trip and the maximum takeoff gross weight of the aircraft, and applies to international flights only [130]. ATA categorised this cost as an indirect cost, while the NASA formula for its determination is:

$$C_{nav} = 0.2 \times 500 \times \sqrt{\frac{m_{to}}{1000}} \tag{5.86}$$

Note: m_{to} is in pounds (*lbs*).

Navigation fees can be calculated by using the AEA formula as:

$$C_{nav} = 0.5 \times \frac{s_l}{1000} \times \sqrt{\frac{m_{to}}{50000}} \tag{5.87}$$

5.4.1.10 Ground handling fee

This cost is included in the DOC in AEA methodology only using the following formula:

$$C_{grd} = 0.1 \times m_{pay} \tag{5.88}$$

Direct Operating Costs:

The total DOC per flight for the ATA methodology for the components defined before is:

$$DOC/flight = C_{dp} + C_{ins} + C_{fc} + C_{fuel} + C_{maint} \tag{5.89}$$

For the NASA methodology:

$$DOC/flight = C_{dp} + C_{ins} + C_{fc} + C_{fuel} + C_{maint} + C_{int} + C_{cc} + C_{lf} + C_{nav} \quad (5.90)$$

For the AEA methodology:

$$DOC/flight = C_{dp} + C_{ins} + C_{fc} + C_{fuel} + C_{maint} + C_{int} + C_{cc} + C_{lf} + C_{nav} + C_{grd} \quad (5.91)$$

The DOC calculations above for the three different methodologies differ in the formulation, the AEA and NASA methodologies differ by the ground handling charge, but produce very different results, due to the empirical formulation of the various formulae.

5.4.2 Case study

To make a good comparison between ATA, NASA, and AEA methodologies, all of them have been applied to the current Airbus and Boeing aircraft. At a glance, the AEA methodology gives the highest DOC values and in turn the highest SMC values as shown in **Figure 5-12** and **Figure 5-13**, respectively. To understand the differences one needs to break down the DOC into its main components and investigate each one.

The first component of the DOC under consideration is the standing charge (or so-called ownership) which consists of depreciation, insurance, and interest. These costs form 30-40% of the DOC and depend mainly on the annual utilisation of the aircraft. Obviously, as the utilisation increases, the standing charges decrease. Although NASA methodology has the lowest average value, the main drawback of NASA methodology is the ambiguous definitions of ranges (short, medium, and long). On the other hand, if the aircraft is fully owned, interest is not included.

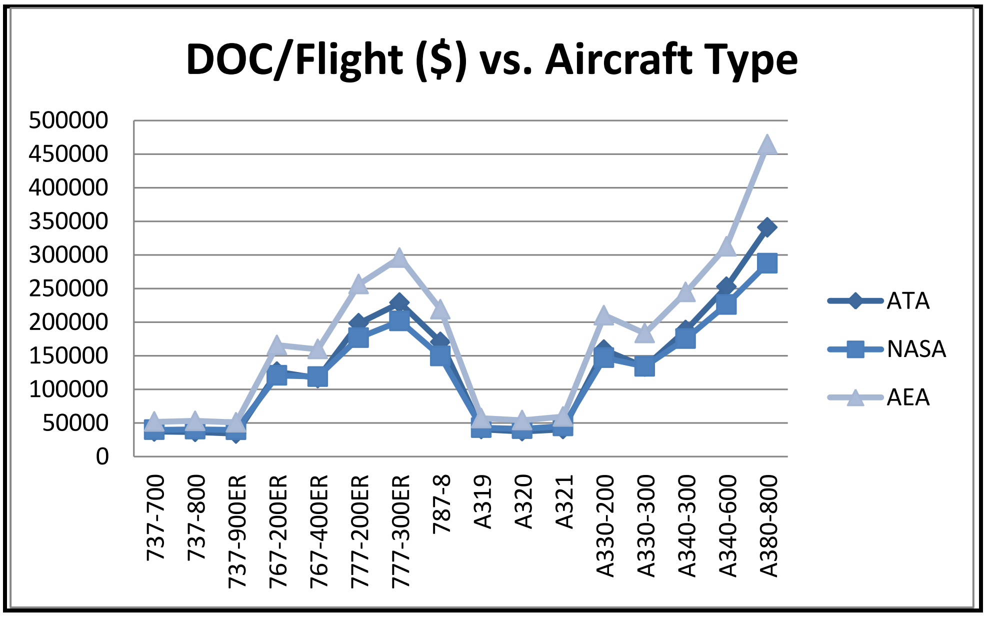

Figure 5-12: DOC for ATA, NASA, and AEA methodologies

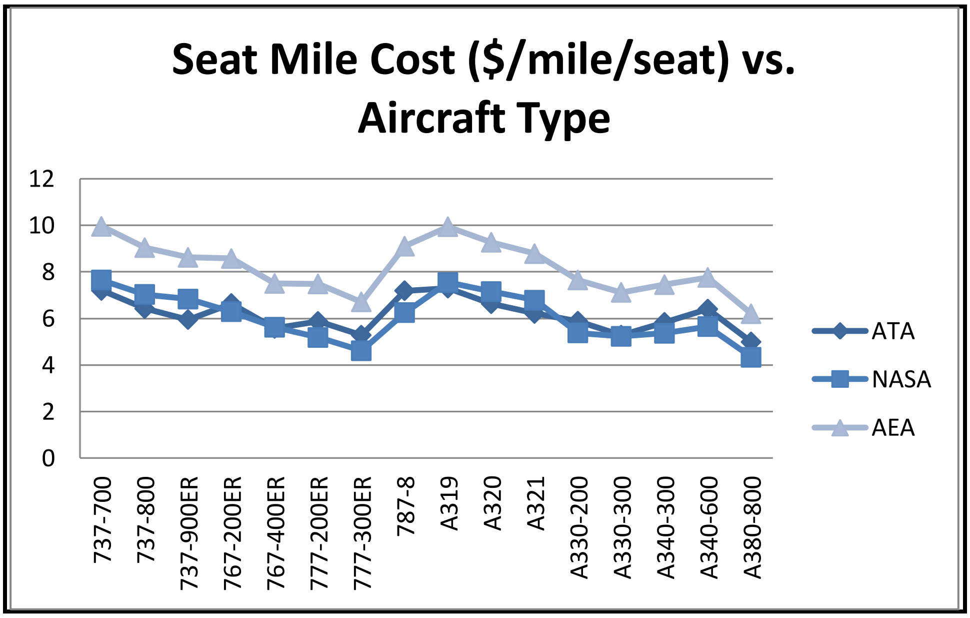

Figure 5-13: Seat/Mile-Cost for ATA, NASA, and AEA methodologies

From **Figure 5-14**, the differences between ATA and NASA are relatively small and hence, ATA methodology is the better choice. From the engineering design point of view, Swan [137] suggests a simple way to overcome this problem by considering a monthly lease cost for new aircraft at about 0.8-0.9% of the aircraft price.

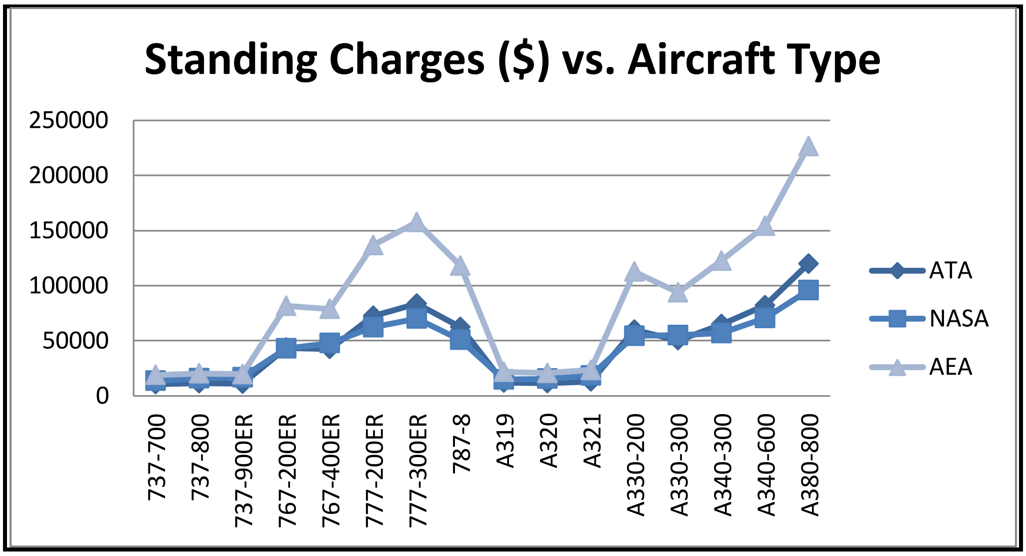

Figure 5-14: Standing charges cost for ATA, NASA, and AEA methodologies

Maintenance cost is the second component that must be considered. In general, it makes up 13% of the DOC [137]. It is based on the utility of the aircraft which is in "steady-state maintenance". That means the maintenance savings of the first five years for new designed aircraft have been finished and the second half-life maintenance cycles has been initiated. Although the most expensive inspections occur once every 3-4 years, the average cost is generally arrived at by rule of thumb**.** **Figure 5-15** shows that the maintenance cost forms 20-25% of DOC for ATA, 8% for NASA, and less than 1% for AEA. These huge differences make the comparison meaningless.

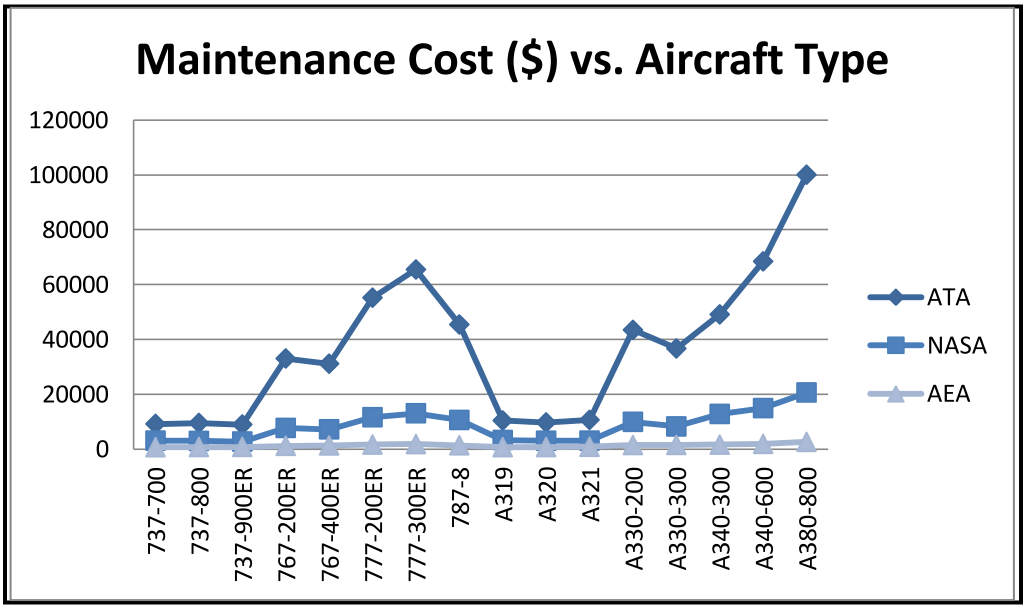

Figure 5-15: Maintenance cost for ATA, NASA, and AEA methodologies

Flight crew cost is another major component of DOC. It is based on both flight time and maximum take-off weight for ATA and NASA, while it is based only on flight time for the AEA methodology. Although there is not much difference between the ATA and AEA results, as shown in **Figure 5-16**, MIT [133] data agree completely with ATA.

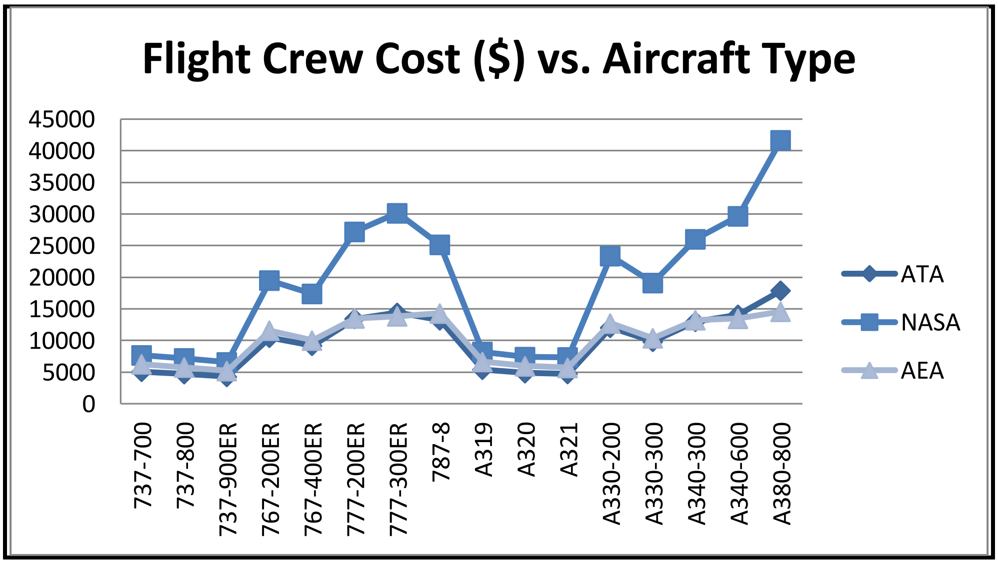

Figure 5-16: Flight crew cost for ATA, NASA, and AEA methodologies

Fuel cost has changed rapidly in the last 10 years and forms a significant parameter that affects the aviation market. There is no difference noticed between the three methodologies as shown in **Figure 5-17**. Although ATA added the oil used cost to the fuel cost, it makes only a very small difference, and hence can be discounted.

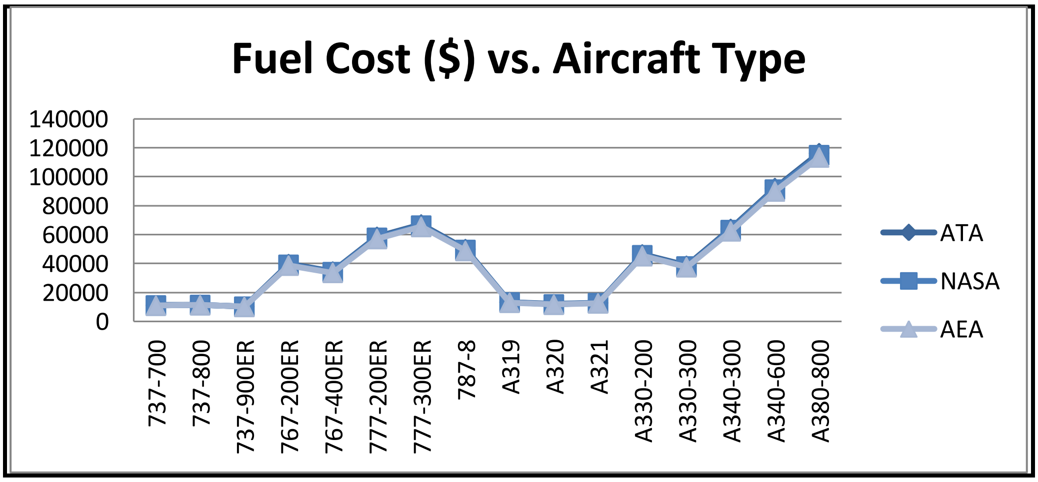

Figure 5-17: Fuel cost for ATA, NASA, and AEA methodologies

Another methodology, which has been developed by Swan [137], was applied. It evaluates the DOC as a function of stage length and seat capacity. It is based on years 1999-2001 data, and one needs to apply an inflation factor of 1.266 to update data to year 2010. **Figure 5-18** shows that Swan methodology gives approximately the same average difference when compared with the ATA methodology. ATA developed statistical equations, which includes attendant crew, landing fee, navigation costs, and other costs associated with operating an aircraft, for estimating IOC. By adding IOC to DOC, ATA methodology has the ability to calculate the total operation cost (TOC) rather than DOC alone.

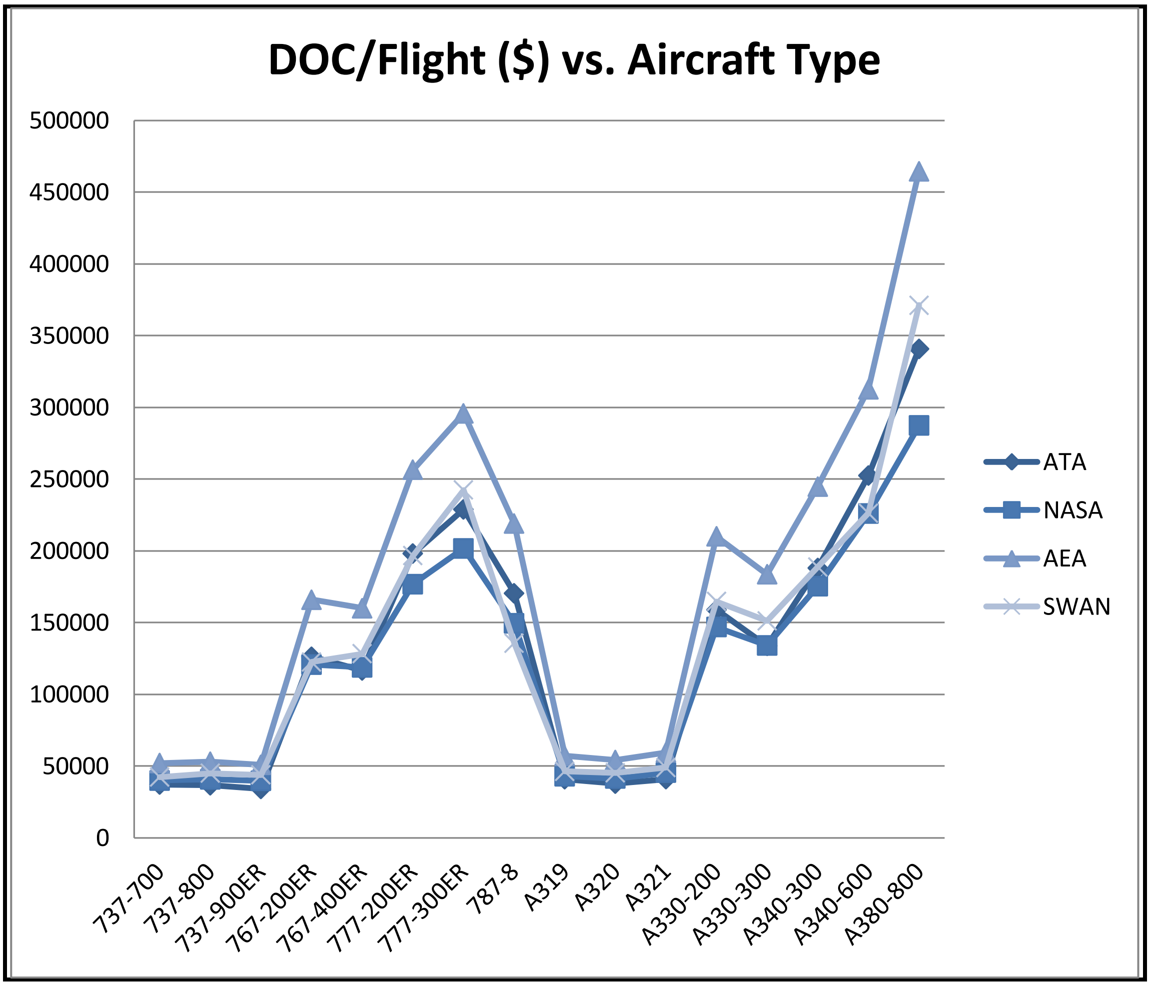

Figure 5-18: DOC for ATA, NASA, AEA, and Swan methodologies

5.5 Conclusions

In the preliminary design phase of the interactive design process, where the maximum takeoff weight needs to be broken down into its components and sub-components, the prediction methodologies become complicated. Estimation methods in this phase generally improve the weight estimation accuracy from 10-15% (in the conceptual design phase) to 5-10%. A modified module has been developed to increase the accuracy to better than 4%. Its output results agree very favourably with the published data of current Airbus and Boeing aircraft.

Cost-benefit analysis is a significant parameter in evaluating competitive aircraft designs. DOC is a very useful and widely-used parameter for comparative analysis. ATA and NASA are two common methodologies that are added to the AEA methodology, allowing the cost estimation by any method the user chooses. All cost estimation methods have been applied to estimate the DOC and SMC for the existing transport aircraft. The results show that ATA and NASA methodologies are close to each other.

The weight estimation was based on the lowest value from one of the many empirical equations. However, in the case of cost analysis no such choice is made. It is up to the student to choose one of the costing methods depending upon operational and fiscal requirements.

Chapter Six

Case Studies

6.1 Introduction

Aircraft design is a complex and iterative process. It is a classical problem based learning Problem-Based Learning scenario. As depicted by figure 1.3, there is a substantial amount of self directed learning in the various stages of design. For students of aerospace engineering studying aircraft design, one or two semesters of study are not enough to understand all the aspects of design methodologies required in the various stages of design. A teaching software tool can greatly assist in understanding and speeding up the design process. It can also provide them with baseline results that can be used to check their analytical calculations against. Students with access to software tools such as the one developed in this research, can experiment with design changes quickly and more efficiently which will enable enhanced and deeper understanding. These benefits and others were stated in chapter one. Understanding and assessing these benefits require solving a practical design problem as done by a typical student team, typically in the final year of an aeronautical engineering degree course, whilst studying aircraft design.

6.2 Aircraft design problems

In this section, two case studies are presented that explore the limitations of conventional undergraduate aircraft design approach and how the software can enhance the student experience to investigate the effect of design variables on the whole aircraft design. The studies are presented as an emulation of a typical student design process.

6.2.1 Case study A

The design process starts with design requirements given to a student team. For the purposes of illustration a typical specification will be assumed. A typical brief comprises the following requirements, but may have other aspects specified as well:

1. A civil aircraft that transports 150 passengers over a range of 5500 km.
2. Maximum cruise speed Mach 0.84 and ceiling above 9500 m.
3. Maximum distance for takeoff to be 2500m, for landing not more than 1500 m.
4. Pressurised cabin with air conditioning and oxygen supply.

According to the above requirements, the student team starts the first stage in the conceptual design phase with the "estimation of maximum takeoff weight". The common estimation approach is to break down this weight into its components, i.e.:

$$m_{to} = m_{oe} + m_{fuel} + m_{pay} \quad (6.1)$$

The operating empty weight (m_{oe}) is:

$$m_{oe} = m_e + m_{crew} \quad (6.2)$$

The procedure of estimation starts by initially guessing a likely value of maximum takeoff weight based on similar aircraft in the market. The available aircraft that match the requirements are:

1. Airbus A319-100, 156 passengers, 6700 km, and 75500 kg.
2. Boeing 737-700, 149 passengers, 6230 km, and 70305 kg.
3. McDonnell Douglas MD-87, 139 passengers, 4300 km, and 63500 kg

If the number of passenger versus aircraft weight data above is plotted, a simple linear equation can be fitted through the data, from which an aircraft with 150 seats is likely to weigh 71000 kg. Students at this stage may decide on a lower weight, due to use of composite construction, so may choose 68000kg as their target weight. This is a reasonable estimate for the class of aircraft under consideration. The next step is to estimate the fuel weight by using the fuel-fraction method. In this method, the aircraft mission is broken down into a number of segments. These segments are: 1- engine warm-up, 2- taxi, 3- takeoff, 4- climb, 5- cruise, 6- loiter, 7- descent, and 8- landing, taxi, and shut down. Fuel weight in each segment is evaluated either by simple calculations or by estimation based on the past experience, or as in the textbooks. Segments 1, 2, 3, 4, 7, and 8 are estimated as: 0.99, 0.99, 0.995, 0.98, 0.99, and 0.992, respectively. Fuel fraction for cruise segment 0.805 is calculated from Breguet's range equation with cruise velocity of 254 m/s (Mach 0.84). Then, total fuel fraction ($\frac{m_{fuel}}{m_{to}}$) is 0.244. Payload weight is determined based on FAR 25 as in chapter three, i.e.:

$$m_{pay} = 90.72 \times N_{pas} = 14058\ kg \tag{6.3}$$

Crew weight, which includes two flight-crew and five flight-attendants, is determined also as in chapter three, or using CS25, i.e.:

$$m_{fl_{crew}} = 93 \times N_{fl_{crew}} = 186\ kg \tag{6.4}$$

$$m_{fl_{att}} = 68 \times N_{fl_{att}} = 340\ kg \tag{6.5}$$

$$m_{op\_it} = 8.617 \times N_{pas} = 1293\ kg \tag{6.6}$$

Hence,

$$m_{crew} = m_{fl_{crew}} + m_{fl_{att}} + m_{op\_it} = 1819\ kg \tag{6.7}$$

Rewriting Equation (6.1) and using Equation (6.2) yields:

$$m_{to} = \frac{m_{crew}+m_{pay}}{1-\frac{m_e}{m_{to}}-\frac{m_{fuel}}{m_{to}}} \tag{6.8}$$

Step three is applying Equation (6.3), using the guess value of ($m_{to} = 68000$) , to calculate empty weight fraction ($\frac{m_e}{m_{to}}$), i.e.:

$$\frac{m_e}{m_{to}} = 0.4975 \text{ , and } m_e = 33830\ kg \tag{6.9}$$

Step four is applying the following Equation to calculate (m_{to}):

$$\frac{m_e}{m_{to}} = 0.97 \times m_{to}^{-0.06} \tag{6.10}$$

, which yields: $m_{to} = 65043\ kg$

Now, a guess must be made of another value for m_{to}, and, step three and four repeated until the guessed value equals the calculated value as in **Table 6-1**:

$\boldsymbol{m_{to}}$ guess	$\boldsymbol{\frac{m_e}{m_{to}}}$	$\boldsymbol{m_e}$	$\boldsymbol{m_{to}}$ calculated
68000	0.4975	33830	65043
65000	0.4989	32429	65418
66000	0.4984	32894	65290
65500	0.4987	32665	65351

Table 6-1: MTOW estimate iterations

The final values are:

Maximum takeoff weight = 65400 kg

Empty weight = 32615 kg

Fuel weight = 16870 kg

The second stage of estimations, which also is based on past experience, includes the evaluation of wing loading, takeoff thrust, and maximum lift coefficient. These estimations are necessary to meet performance requirements such as takeoff field length, landing field length, and climb rate. Selecting wing loading = 500 kg/m^2, gives wing area = 130.8 m^2, while selecting thrust loading equal to 0.35, gives takeoff thrust

= 50163 pound (lbs). A point to be noted here is that selecting the highest possible wing loading and the lowest possible thrust loading (which achieves all performance requirements) results in an aircraft with lowest weight and lowest cost. High aspect ratio decreases induced drag in cruise and saves fuel. Therefore, by selecting aspect ratio = 10, the following aerodynamic coefficients are determined:

Zero-lift drag = 0.0264, maximum lift coefficient = 1.6, takeoff lift = 2.1 coefficient, and landing lift coefficient = 2.4.

At this point, the student team is still working together to select the shape (configuration) of the proposed aircraft. A conventional configuration is selected based on similarly available aircraft in the market which are characterised as:

1. Most of them have a low wing configuration.
2. Most of them have wing-mounted engine, some have rear fuselage-mounted engine.
3. All have a tricycle layout, are wing-mounted, and have retractable into wing-fuselage intersection landing gear.
4. A conventional tail is more used than a T-tail configuration.
5. The database and experience dealing with conventional configuration is very large. It is limited and even non-existent in some of the other configuration.

After the aircraft's shape selection, each student starts his/her component design as follows:

Fuselage design: fuselage shape depends mainly on seat configuration which in turn depends on the number of passengers. It can be described by two parameters; fineness ratio and tail-cone length to fuselage diameter ratio. Past experience for transport aircraft shows that the fineness ratio's range is 7-11 and 2.6-4 for the second ratio. Seat configuration includes the number of seats abreast, number of aisles, and seat specifications (e.g. width, pitch, height, etc). For 150 passengers, selection of 6 seats abreast and single aisle, gives 25 rows, seat pitch = 0.91 m, cabin length = 22.5 m, and fuselage diameter = 4 m. Selecting nose cone length = 4 m, and rear cone ratio = 3. Hence, total fuselage length = 38.5 m.

Engine selection: since total takeoff thrust required is 50163 pounds, with number of engines = 2, thrust /engine = 25082 pounds. Hence, CFM 56-5A1 is selected with thrust = 25100 pounds, and 2266 kg dry weight. It is decided to mount the engines under the wing, forward to centre of gravity (the aircraft works as a tractor).

Wing design: this is the most important component of an aircraft, since its geometry affects all other aircraft components. Some wing variables are determined which are area, aspect ratio, and wing configuration. Others must be selected or evaluated. A sweep angle of 30 degrees is selected to achieve a cruise speed of M0.84. The second most important wing variable, after wing area, is its aerofoil. This is responsible for the generation of the optimum pressure distribution on wing surfaces (top and bottom surfaces) such that the required lift is created. The aerofoil selection is based on its general features. These features include the maximum lift coefficient, highest lift curve slope, and lowest pitching moment coefficient. An aerofoil NACA 64A413/411 is selected with thickness ratio at root = 0.13 and at tip = 0.11. Taper ratio is estimated equal to 0.3, since it has a great effect on lift distribution calculations, while the incidence angle is estimated = 2 degrees. Finally, flaps are used to increase exposed wing area which means a further increase in lift. Flap design variables are estimated as a fraction of either root chord or semi-span of the wing. The inboard span fraction is estimated = 0.5, while outboard span fraction is = 0.8. The inboard and the outboard chord fractions are estimated = 0.2.

Empennage design: The next step is placing the empennage on the aircraft to determine the empennage moment arms. These moment arms are obtained from the proposed aircraft drawing. In order to keep aircraft weight and drag as low as possible, it is desirable to keep tail area as small as possible. Tail area is determined by estimating tail volume coefficients based on past experience. A value of 0.8 and 0.06 are selected for horizontal and vertical tail volume coefficients, respectively. **Table 6-2**

shows the tail design variable estimations. Again, these estimations are based on similar available aircraft in the market and the past experience.

Design variable	Horizontal tail	Vertical tail
Aspect ratio	5	1.8
Sweep angle	30	40
Taper ratio	0.3	0.3
Thickness ratio	0.12	0.13
Aerofoil	NACA 0012	NACA 0015

Table 6-2: Tail design variable estimations

At this point the conceptual design phase is finished and the preliminary stage starts. In preliminary design phase, students apply semi-empirical formulae to calculate the parameters of aircraft components needed for detailed design phase in a higher accuracy. Starting with the Geometry section, **Table 6-3** and **Table 6-4** show the calculated data for each aircraft component based on the formulae presented in chapter three:

Wing:

Output Parameter	Value
Span	36.17 m
Root (centreline) Chord	5.56 m
Tip Chord	1.67 m
Mean Geometric Chord	3.58 m
Mean Aerodynamic Chord	3.97 m
Exposed Wing Span	31.78 m
Exposed Wing Area	108 m^2
Distance from Fuselage Nose to Wing Quarter Root	14.47 m
Distance from Fuselage Nose to Wing Centre of Pressure	18.31 m

Table 6-3: Wing geometry output parameters

Tail:

Output Parameter	Horizontal Tail	Vertical Tail
Area	21.66 m^2	14.92 m^2
Span	10.41 m	5.18 m
Root Chord	3.20 m	4.45 m
Mean Geometric Chord	2.08 m	2.88 m
Tail Arm	19.16 m	19.02 m

Table 6-4: Tail geometry output parameters

Fuselage: Wetted Area = 433.54 m^2. Nacelle: Wetted Area = 14.93 m^2. For Weight section, gust load factor = 6.288 and manoeuvre load factor = 4.125. **Table 6-5** shows the calculated aircraft component weights:

Component	Weight (kg)
Wing	9411.2
Fuselage	9788.6
Tail	1251.9
Propulsion (engines + nacelles)	7370.8
Undercarriage	2650.2
Surface Controls	1077.5
Systems	4769.7
Furnishings	3593.1
Empty Weight	39913.0
Operating Items	1292.5
Crew	526
Operating Empty Weight	41731.5
Payload	18030.5
Zero-Fuel Weight	59762
Fuel	17000
Maximum Takeoff Weight	76762

Table 6-5: Calculated aircraft component weights

It should be noted that the estimated weight is higher than the target weight. In the CG and Stability section, the moments of each component are calculated to evaluate CG output parameters as in **Table 6-6**:

Output Parameter	Value
C.G. Position of Empty Aircraft	18.23 m , from Fuselage Nose
Aft Limit	18.45 m , from Fuselage Nose
Forward Limit	17.30 m , from Fuselage Nose
Neutral Point	19.23 m , from Fuselage Nose
Static Margin	0.47

Table 6-6: CG output parameters

In the Aerodynamics section, zero-lift drag coefficients, for each aircraft component, are computed. The results are in **Table 6-7**:

Component	Zero-lift Coefficient
Exposed Wing	0.00655
Fuselage	0.00800
Tail	0.00235
Nacelles	0.00906
Interference	0.00050

Table 6-7: Zero-lift drag coefficients for aircraft components

The total zero-lift drag coefficient = 0.0264.

The total drag (zero-lift + induced) coefficient for a given lift coefficient (Polar Plot) is computed as in **Table 6-8**:

CL	CD
0.0	0.02717
0.2	0.02828
0.4	0.03270
0.6	0.04041
0.8	0.05143
1.0	0.06575
1.2	0.08337
1.4	0.10429
1.6	0.12852

Table 6-8: Lift-drag coefficients

The takeoff lift coefficient for trimmed aircraft with 20 degrees flap deflection is computed = 1.97, and landing lift coefficient for trimmed aircraft with 40 degrees is computed = 2.23.

For the Performance section, **Table 6-9** shows the calculated output parameters:

Performance Parameter	Value
Rate of Climb at Start	14.97 m/s
Rate of Climb at End	6.45 m/s
Rate of Descent at Start	-13.56 m/s
Rate of Descent at End	-10.29 m/s
Balanced Field Length	2657 m
Landing Field Length	1238 m
Takeoff Stall Speed	69.2 m/s
Landing Stall Speed	58.5 m/s

Table 6-9: Performance output parameters

Finally, the Cost estimation section takes place. The aircraft cost is computed based on its empty weight plus engine price. In year 2010, most airliners are seen to cost $750 to $800 per pound. Then, the airframe cost = $68.58 million and total engine cost = $10.28 million. Therefore, aircraft cost = $78.85 million.

Direct operating cost (DOC) is computed as in **Table 6-10**:

DOC Component	Cost ($)
Depreciation	10164.7
Insurance	702.1
Interest	8380.2
Maintenance	1701.5
Fuel	10047.8
Flight Crew	5356.2
Cabin Crew	2001.6
Landing Fees	1052.9
Navigation Fee	5991.9
Ground-Handling Charges	3170.7

Table 6-10: DOC component costs

The total DOC = $48569.5, DOC/nm = $16.4, and SMC = ¢10.9.

At this point the preliminary design phase is finished. The student team may use a simulator to assess their design, especially aspects of dynamic stability. It may seem to be a good design even in terms of dynamic stability. The above process assumed that students are able to determine the appropriate initial estimates of the required data, and if they were chanceful the estimates, they may end up believing that aircraft design is simple. On closer examination in a preliminary design review, the design may be taken apart by the lecturers. The question is: Why is the design considered to be bad if the aircraft can fly and in stable condition? If so what design improvements are necessary? The next section will answer these questions in some detail.

6.2.1.1 Design assessment and discussion

Many reasons can be attributed to the failure. Simply, it does not meet the requirements. First of all, the proposed aircraft cost (\$78.85 million) is much higher than the cost of available aircraft in the market. From the market survey, prices for the Airbus 319 = \$74.7 million and for the Boeing 737-700 = \$67.9 million. Hence, the cost must be reduced to lower than \$68 million for it to become a competitive aircraft. Reducing the cost means reducing its weight. In conceptual design, maximum takeoff weight was 65400 kg, while it was estimated to be 76762 kg in the preliminary design phase, as compared to the initial guess of 68000kg. So, the student team must reduce this weight. They start again to redesign the aircraft and this is actually when the iterative nature of aircraft design becomes evident. They may spend two weeks for the first iteration. One benefit of using a preliminary software tool is to speed up the design process. By using such a tool, a single iteration takes just few seconds to perform the preliminary design process.

The objective of the second and subsequent iterations is all about decreasing the aircraft weight. For students, who are doing aircraft design for the first time, they have little idea on how to select, what is the available range of variation, and what are the most significant design variables. For design engineers in industry, working in the conceptual design section the selection process is a matter of experience and they are able to identify the key parameters that will allow the specifications to be met. Since the fundamental tool is “selection” in conceptual design, the student team will try to reselect all the design variables. In the first iteration, the aircraft’s weight increased by 20%. In the second iteration, an attempt will be made reduce any design variable (if applicable) by 10% at least. For the wing component, wing loading is increased from 500 kg/m^2 to 550 kg/m^2 in order to reduce wing area from 130.8 m^2 to 118.9 m^2, decrease the aspect ratio from 10 to 9, and so on. Due to the number of passengers’ requirement, there is a narrow band of variations in fuselage design variables. For example, fuselage length is reduced by reducing seat pitch from 91.4 cm to 84.4 cm, and/or reduce fuselage diameter from 4 m to 3.8 m by reducing seat width (although, these changes affect the passengers' comfort). Also, reducing the thrust loading from 0.35 to 0.316 means a reduction in engine thrust and in turn in total fuel burn. This leads to the reduction of fuel weight from 17000 kg to 16000 kg. The tail is used mainly for stability and control and its weight is relatively small with respect to wing or fuselage, so they do not need to make any changes to the selected values. At the end of the second iteration, aircraft weight is reduced to 71372 kg, and its cost is now $68.7 million.

Although the second iteration improved aircraft cost and weight, it does not comply with one of the other requirements: which is the Balanced Field Length (BFL). The BFL increased from 2682 m (in the first iteration) to 2843 m (in the second iteration). Utilising Torenbeek's equation for BFL, many parameters have an effect on BFL such as: aircraft weight, starting climb velocity (v_2), and engine thrust. There are limitations on reducing aircraft weight any more due to the fuselage weight limitation (which is based on seat configuration, i.e. cabin length and diameter). Also, reducing aircraft weight, which implies a reduction in engine weight and in turn engine thrust, conflicts with shortened BFL (which requires an increase in engine thrust). *"Analysis of the takeoff characteristics is the most complicated item of performance; this applies particularly to transport aircraft, where the possibility of engine failure must be considered"* [10]. The student team can find itself in a very awkward situation. They may not know which design variable should be changed to satisfy BFL requirement. The solution is simplified by using the preliminary design software. Since the software is able to perform all the necessary computations very quickly, it allows the student to experiment by varying the key parameters and to study the effect of the change. The parametric studies module in this software tool facilitates this process. Applying most (or even all) design variables to see which of them has a great effect on BFL. Learning also how much they should vary the design variable. Typically, increasing wing area from 118.9 m^2 to 122 m^2, reducing wing sweep angle from 27 degrees to 25 degrees, and increasing takeoff flap angle from 20 degrees to 23 degrees, will shorten BFL by more than 300 m (which meets the requirement). This is the major benefit of the software tool which helps the students to understand and analyse the effect of each design variable on the whole design as well as on a specific parameter and how one parameter affects the others.

After these iterations and improvements, the student team will port the design variables to a flight simulator to investigate the performance and stability of the designed aircraft. Unfortunately, the aircraft is laterally unstable as shown in the **Figure 6-1.**

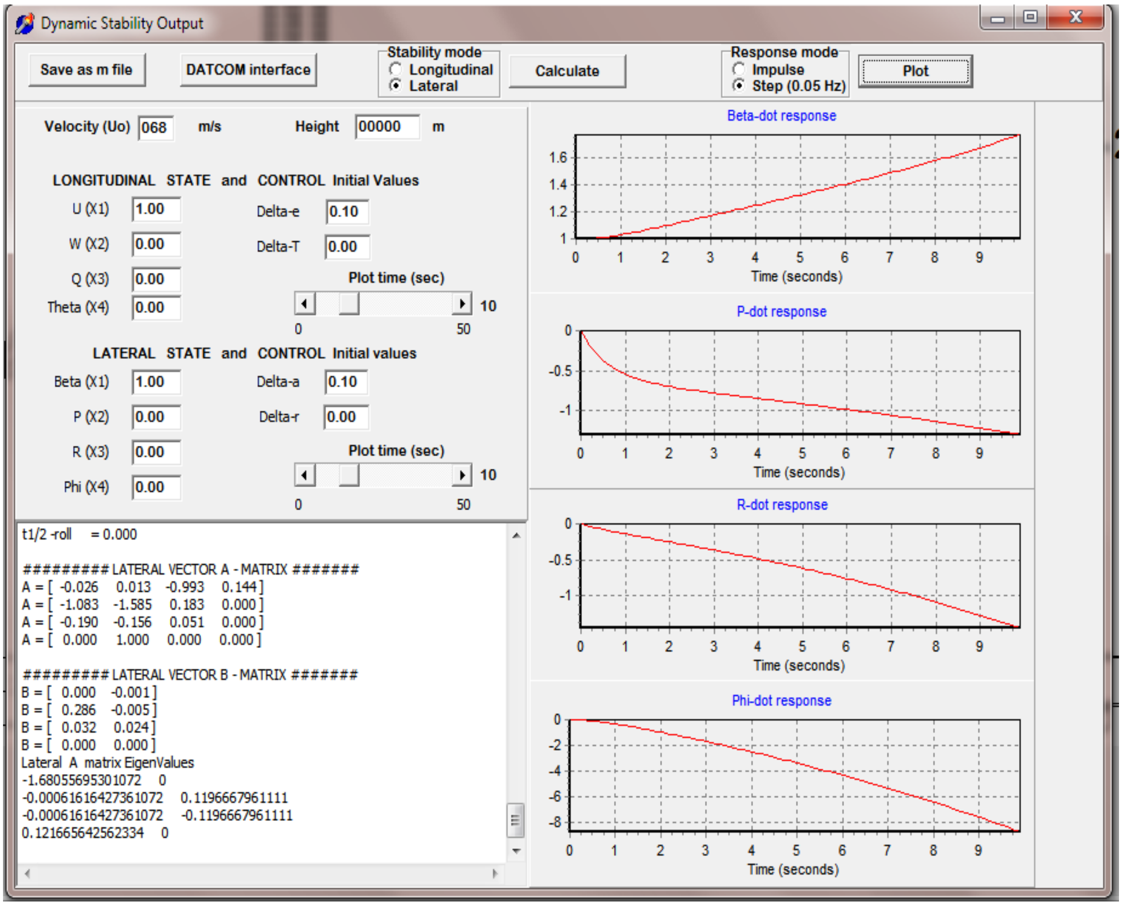

Figure 6-1: Lateral Aircraft Response to a Step aileron deflection

From **Figure 6-1** it can be noted that one of the eigenvalues of the lateral dynamics has a positive root, associated with either spiral or roll mode. The above dynamic analysis is performed using the software. Time is saved due to the fact that the students do not need to port their designs on to a simulator, only to find it dynamically unstable. Students only need to run the flight simulations when all design requirements are satisfied and the stability is indicated. These simulations can be performed either on a flight simulator if available or can be done by using one of the freeware simulation packages such as X-Plane [138] or Flight-gear [139].

In the present context, lateral stability can be achieved by increasing the volume coefficient of the vertical tail from 0.06 to 0.07. This increment will increase aircraft weight by less than 300 kg, which is a small increase with respect to the whole aircraft weight. The availability of the dynamic stability module in one package tool is another benefit that facilitates design process and saves time and effort of the students who may only have one or two months to analyse and modify their design. Due to many iterations and repeated hand calculations, students may make uncertain mistakes which in turn gives wrong results and may cause the whole design to fail. The additional benefit of using the software tool is that it offers error-free calculations. It needs to be stressed here that, students need to perform all the necessary computations using the long hand methods, so that they become familiar with the underlying theories. The iADS software can then be used to validate their design. Or as a tool to refine their initial design that they arrived at by using the long hand computational method. These results can be used by the students as a benchmark, against which they can check and compare their long hand calculations.

Appendix II shows the full text output results of the software for the designed aircraft. It should be noted that the calculated weight (70149 kg) is lower than that calculated by the students, since they used Torenbeek's formulae. This accurate weight estimation is another benefit that supports students in evaluating their design. Hence, the MTOW is reduced the aircraft cost is reduced also, (lower than Boeing prices) and the design now is actually a competitive one. The ability of providing the input file for DATCOM software as in Appendix III for this case study is that it allows students to use DATCOM for computing the stability and control derivatives. Appendix IV shows the '.m' file of the designed aircraft produced by the software that helps students to use it in MATLAB software for complete dynamic simulation, or an engineering flight simulator if available.

A detailed examination of the student design and that of an existing aircraft is presented in the following table. **Table 6-11** consists of two sections, requirements and output. Only some output variables are shown to emphasise the accuracy and validity of software when compared to published data of a similar production aircraft.

Requirements	
No. of Passengers	150
Range	5500 km
Max. Cruise Speed	0.84M
BFL	< 2500 m
LFL	< 1500 m

Key Parameters	Designed Aircraft	Boeing 737-700
MTOW (kg)	70150	70305
OEW (kg)	36119	38147
Total Fuel Burn (kg)	13523	15100
Max. Cruise Speed	0.84M	0.78M
Flight Time (hr:min)	5:57	6:58
BFL (m)	2421	2200
LFL (m)	1208	1375
Thrust/Engine (lbs)	22770	26300
Range (km)	5500	6230
No. of Passengers	150	149
Overall Length (m)	36.15	32.18
Overall Width (m)	3.80	3.89
Aircraft Price ($)	65,800,000	67,900,000

Table 6-11: Comparison between the designed aircraft and Boeing 737-700

6.2.2 Case study B

The second case study is for a light business jet aircraft. The design process is similar to the first case study. The design process starts with design requirements given to the student team. These requirements include the following:

1. A small business aircraft that transports 4-6 passengers over a range of 1500nm with reserve of 100nm.
2. Turbo-fan engines and maximum speed M0.7.
3. BFL less than 1000 m.

According to the above requirements, the student team starts the first stage in the conceptual design phase which is the “estimation of MTOW”.

The procedure of estimation starts by initially guessing a likely value of MTOW based on similar aircraft in the market. The available aircraft that match the requirements are:

1. Adam A700, 4-6 passengers, range 1430nm, and MTOW = 4250 kg.
2. Cessna 525 (CJ1+), 3-6 passengers, range 1300nm, and MTOW = 4853 kg.
3. Cessna 525 (CJ4), 4-9 passengers, range 1900nm, and MTOW = 7671 kg.

The problem is similar to the first one. So, 5000 kg is a reasonable assumption for the MTOW. The next step is to estimate the fuel weight by using the fuel-fraction method. In this method, the aircraft mission is broken down into a number of segments. These segments are: 1- engine warm-up, taxi, and takeoff, 2- climb, 3- cruise, 4- descent, landing, taxi, and shut down. The fuel weight in each segment is evaluated either by simple calculations or by an estimation based on the past experience as in the textbooks. Segments 1, 2, and 4 are estimated as: 0.97, 0.985, and 0.995, respectively. The fuel cruise segment is 0.858, calculated from Breguet’s range equation with cruise velocity of 211 m/s. Then, the total fuel fraction ($\frac{m_{fuel}}{m_{to}}$) is 0.202. Payload weight is determined based on FAR 25 as 90 kg per person plus a useful load of 30 kg per person, i.e.:

Using Equation (6.3) with $N_{pas} = 6$, $m_{pay} = 720\ kg$

Using Equation (6.4) and with crew compliment of two, Crew weight,
$m_{fl_{crew}} = 186\ kg$

The common estimation approach is to break down the MTOW into its components as defined by Equation (6.1). The OEW was defined by Equation (6.2) and the empty mass fraction was defined earlier as Equation (6.8).

Finally application of the following equation with empty mass fraction determined by Equation (6.8) which was 0.6168:

$$\frac{m_e}{m_{to}} = 1.47 \times m_{to}^{-0.1} \tag{6.11}$$

, which yields: $m_{to} = 5912\ kg$

The initial guess for the MTOW was 5000kg, and the calculated value turns out to be 5912kg. It is necessary to guess another value of MTOW and the above process is repeated till the calculated value is more or less the same as the initial guess.

Table 6-12 shows the iterative nature of this process:

m_{to} guess	$\frac{m_e}{m_{to}}$	m_e	m_{to} calculated
5000	0.6168	3084	5912
6000	0.647	3882	3665
5500	0.6333	3483	4542
5300	0.627	3323	5013
5200	0.6238	3244	5284
5250	0.6254	3283	5145
5225	0.6246	3264	5214

Table 6-12: MTOW estimate iterations

So the MTOW is computed to be 5225kg, which appears reasonable. The Empty weight is calculated to be 3264 kg and the fuel weight to be 1055 kg.

The Cessna citation CJ1+ is a very similar aircraft, but with less range and speed to the one that is being designed. So an increase in the fuel weight and bigger engine to achieve the required range and design speed. It is no surprise that students might choose the tried and tested configuration of a number of light Jet aircraft. So a conventional arrangement, i.e., swept back wings, T-tail and fuselage mounted engines are most likely to be the preferred arrangement. One reason this makes this configuration mandatory rather than preferred is the limited fuselage length due the number of passengers. Another reason is the difficulty of controlling the CG movement if engines are mounted on the wing.

The second stage of estimations includes the evaluation of wing loading and takeoff thrust. These estimations are necessary to meet performance requirements such as BFL, LFL, and climb rate. Selecting wing loading = 150 kg/m^2, gives wing area = 34.84 m^2, while selecting thrust loading equal to 0.4, gives takeoff thrust = 4180 pounds (lbs). Selecting the highest possible wing loading and the lowest possible thrust loading (which achieves all performance requirements) results in an aircraft with lowest weight and lowest cost.

Fuselage design: For 6 passengers, 2 seats abreast with seat pitch = 1.5 m, and 1.5 m for toilet and door, yields cabin length = 6 m and fuselage diameter = 1.8 m. Selecting nose cone length = 4 m, and rear cone ratio = 3. Hence, total fuselage length = 15.4 m.

One of the important key parameters that must be taken into consideration in this case study is the wing position to fuselage length ratio. In a conventional configuration, where the cabin length is large, this ratio is about 0.33-0.35 as in Case A. In light aircraft, where the engines are mounted at the rear of fuselage, this ratio is increased to be 0.44 to balance the aircraft (CG position) and to achieve a good longitudinal stability.

Engine selection: Since total takeoff thrust required is 4180 pounds, with number of engines = 2, thrust /engine = 2090 pounds. Hence, FJ 44-2A is selected with thrust = 2300 pounds, and 240 kg dry weight.

Wing design: Aspect ratio is selected first. Although, high aspect ratio decreases induced drag in cruise and saves fuel, a value of 8 is reasonable due to the limited fuselage length. A sweep angle of 15 degrees is selected to achieve a cruise speed of M0.7. An aerofoil with average thickness ratio = 0.12 is selected. Taper ratio is estimated equal to 0.3, while the incidence angle is estimated = 2 degrees. Finally, the inboard span fraction is estimated = 0.3, while outboard span fraction is = 0.8. The inboard and the outboard chord fractions are estimated = 0.3.

For **Empennage design**, it was decided to use the T-tail configuration. In order to keep aircraft weight and drag as low as possible, it is desirable to keep tail area as small as possible. Tail area is determined by estimating tail volume coefficients based on past experience. A value of 0.6 and 0.05 are selected for horizontal and vertical tail volume coefficients, respectively. **Table 6-13** shows the tail design variable estimations. Again, these estimations are based on Cessna aircraft.

Design variable	Horizontal tail	Vertical tail
Aspect ratio	5	2.5
Sweep angle	15	30
Taper ratio	0.3	0.3
Thickness ratio	0.12	0.12
Volume Coefficient	0.6	0.05
Aerofoil	NACA 0012	NACA 0012

Table 6-13: Tail design variable estimations

At this point the conceptual design phase is finished and the preliminary stage starts. In the preliminary design phase, students apply semi-empirical formulae to calculate the parameters of aircraft components needed for detail design phase in a higher accuracy. Starting with the Geometry section, **Table 6-14** and **Table 6-15** show the calculated data for each aircraft component based on the formulae presented in chapter three:

Wing:

Output Parameter	Value
Span	16.69 m
Root (centreline) Chord	3.21 m
Tip Chord	0.96 m
Mean Aerodynamic Chord	2.29 m
Distance from Fuselage Nose to Wing Quarter Root	7.51 m

Table 6-14: Wing geometry output parameters

Tail:

Output Parameter	Horizontal Tail	Vertical Tail
Area	5.89 m^2	4.08 m^2
Span	5.43 m	3.19 m
Root Chord	1.67 m	1.96 m
Mean Aerodynamic Chord	1.09 m	1.28 m
Tail Arm	8.12 m	7.13 m

Table 6-15: Tail geometry output parameters

Fuselage: Wetted Area = 73.8 m^2. Nacelle: Wetted Area = 4.5 m^2.

For Weight section, **Table 6-16** shows the calculated aircraft component weights:

Component	Weight (kg)
Wing	999.9
Fuselage	1161.2
Tail	227.7
Propulsion (engines + nacelles)	772.7
Undercarriage	325.5
Surface Controls	199.9
Systems	1022.2
Furnishings	386
Empty Weight	5095.1
Operating Items	51.7
Crew	186
Operating Empty Weight	5332.8
Payload	721.2
Zero-Fuel Weight	6054
Fuel	1055
Maximum Takeoff Weight	7109

Table 6-16: Calculated aircraft component weights

In CG and Stability sections, **Table 6-17** shows the computed parameters:

Output Parameter	Value
C.G. Position of Empty Aircraft	8.62 m , from Fuselage Nose
Aft Limit	8.64 m , from Fuselage Nose
Forward Limit	8.37 m , from Fuselage Nose
Neutral Point	8.78 m , from Fuselage Nose
Static Margin	0.16

Table 6-17: CG output parameters

In an Aerodynamics section, zero-lift drag coefficients, for each aircraft component, are computed. The results are in **Table 6-18**:

Component	Zero-lift Coefficient
Exposed Wing	0.0071
Fuselage	0.006
Tail	0.0027
Nacelles	0.0034
Interference	0.0006

Table 6-18: Zero-lift drag coefficients for aircraft components

The total drag (zero-lift + induced) coefficient for a given lift coefficient was computed as in **Table 6-19**:

CL	CD
0.0	0.0208
0.2	0.0220
0.4	0.0270
0.6	0.0358
0.8	0.0485
1.0	0.0650
1.2	0.0853
1.4	0.1095
1.6	0.1375

Table 6-19: Lift-drag coefficients

The takeoff lift coefficient for trimmed aircraft with a 15 degree flap deflection is computed to be 2.12, and landing lift coefficient for trimmed aircraft with 30 degrees is computed to be 2.6.

For the Performance section, **Table 6-20** shows the values of the calculated parameters:

Performance Parameter	Value
Rate of Climb at Start	11.0 m/s
Rate of Climb at End	1.04 m/s
Rate of Descent at Start	-16.95 m/s
Rate of Descent at End	-12.38 m/s
Balanced Field Length	1042 m
Landing Field Length	570 m
Takeoff Stall Speed	39.3 m/s
Landing Stall Speed	33.6 m/s

Table 6-20: Performance output parameters

Finally, the aircraft cost is computed based on its empty weight plus engines price. Based on year 2010 data, the airframe cost works out to be \$7.43 million and engine cost at \$1.35 million each. Therefore, the aircraft cost is \$8.79 million. The total DOC is calculated to be \$7616 and DOC/nm as \$5.1, and SMC to be ¢85.

6.2.2.1 Design assessment and discussion

If the design aim was to compete with a Cessna CJ1+, then the design is a failure, with high weight and high cost, when compared with the MTOW of CJ1+ being 4853 kg and its cost is \$7 million. In conceptual design, maximum takeoff weight was 5225 kg, while it was estimated to be 7109 kg in the preliminary design phase. So the weight has to be reduced.

The objective of the second and subsequent iterations is all about decreasing the aircraft weight. In the first iteration, the aircraft's weight increased by 33%. In the second iteration, a reduction of at least 20% in the MTOW would be necessary. In this case, very light aircraft have a different scenario than that for heavy aircraft (Case A). Any reduction must be considered carefully. For example, students may reduce the fuselage length by reducing seat pitch from 1.5 m to 1.25 m, and/or reducing fuselage diameter from 1.8 m to 1.5 m by reducing seat width (although these changes affect the passengers' comfort). Keep in mind that the fineness ratio must be greater than 7. Hence the total fuselage length is reduced more than 1.5 m. For the wing component, they may reduce wing area from 34.84 m^2 to 27.87 m^2 (20%). Also, reducing engine thrust from 2300 to 2100 pounds. This leads to the reduction of fuel weight from 1055 kg to 555 kg. The tail is used mainly for stability and control and its weight is relatively small with respect to the wing or fuselage, so they do not need to make any change to the selected values. At the end of the second iteration, aircraft weight is reduced to 6116 kg, and its cost is now \$7.3 million. Although the second iteration improved aircraft cost and weight, the BFL still too high, i.e. it is 1047 m. Typically, increasing takeoff flap angle from 15 degrees to 22 degrees only, will shorten the BFL below 1000 m, i.e. 988 m (which meets the requirement). Comparing this value with the published value for the Cessna 525 (which is 972 m), it seems an acceptable value.

The first iteration normally is done using the formulae presented in chapter 3. The process is time consuming, normally the various aspects of design run concurrently and some information may or may not be available. The availability of iADS at an early stage of the design phase allows the effect of parameter variation to be evaluated, and by using representative values of similar aircraft, the design can be finely tuned quickly and efficiently. Appendix II (b) shows the detailed output of the iADS software, after necessary changes have been made to reduce the weight and cost. The initial design did not meet the BFL requirements and by increasing the flap deflection, that too was met. **Table 6-21** shows a comparison between the designed aircraft and Cessna CJ1+ and CJ4:

Key Parameters	Designed Aircraft	Cessna CJ1+	Cessna CJ4
MTOW (kg)	6116	4853	7761
EW (kg)	4602	3069	4500
Range (nm)	1500	1300	1900
Max. Cruise Speed	0.7M	0.6M	0.84M
BFL (m)	988	991	972
Thrust/Engine (lbs)	2100	1965	2300
No. of Passengers	6	3-5	6-9
Overall Length (m)	13.75	12.98	16.26
Overall Width (m)	14.93	14.30	15.49

Table 6-21: Comparison of designed aircraft and published data for CJ1+ & CJ4

From **Table 6-21** it can be seen that the designed aircraft is positioned between the CJ1 and the CJ4. It has a smaller engine than CJ4, hence lower fuel costs. The cruise Mach number is at the designed value. The design is viable, and competitive, each design cycle takes fraction of a second to complete.

This case study shows the ability of the iADS software to design a wide range of aircraft (not just the transport aircraft) with different standard configurations. The accurate weight estimation of iADS allows students to design an actual competitive aircraft. Time is a critical issue for students. Hence the availability of iADS speeds up the design process as well as it teaches them different aspects of aircraft design practically. The designed aircraft's 3-View graphic is presented as **Figure 6-2:**

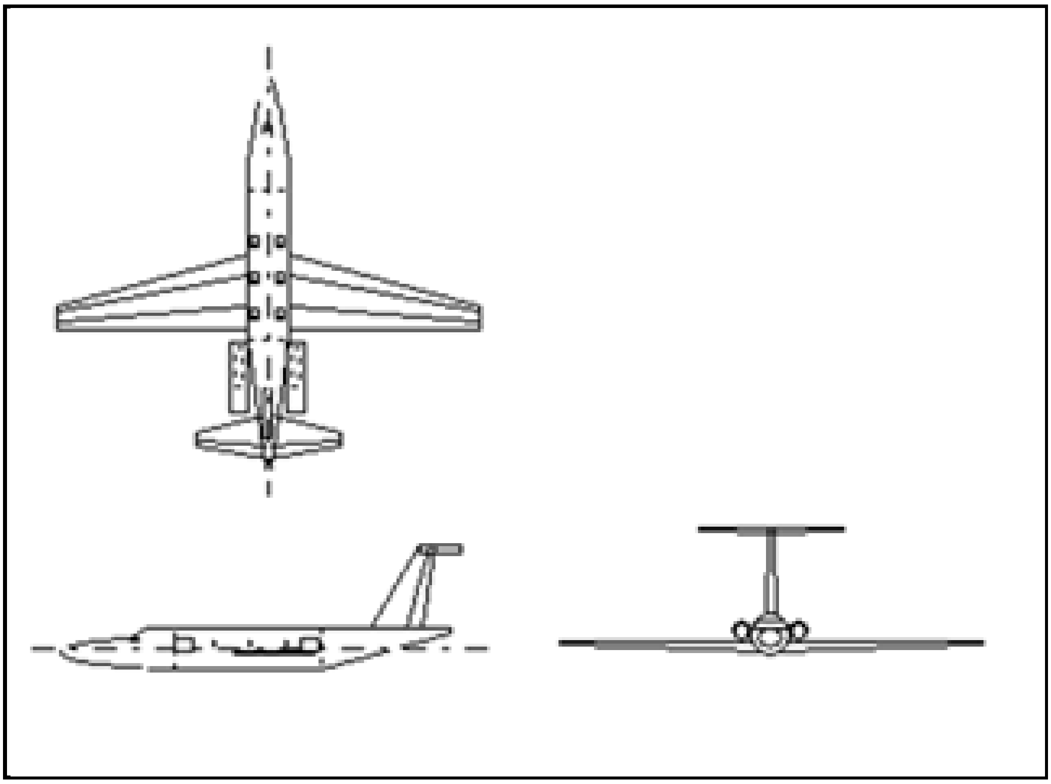

Figure 6-2: 3-View of the designed aircraft

Appendix V shows the published data for Cessna 525, CJ4 [140]. It is easily compared with Appendix II (b), which shows the full text output results of the software for the designed aircraft. For instance, the CJ4 reaches the service ceiling of 45000ft in 29 minutes giving it a climb rate of 1551 ft/min, whereas the designed aircraft achieves a climb rate of 1230 ft/min. The designed aircraft has 400lbs less thrust as compared to the CJ4, hence a slightly inferior climb performance.

Once a likely design candidate has been identified, refinement can be done using the optimiser, and the aircraft could be optimised for either: MTOW, BFL or DOC. As an example, for the same test case it was decided to optimise the design for MTOW, the constraints were set on BFL less than 1000m, and stage length fixed at 1500nm.

Appendix VI shows the typical design with optimiser output. In this case convergence was reached in 2065 iterations.

The DOC has indeed been reduced, and the basic geometry of the aircraft changed. Some of the changes are noted in **Table 6-22**. It is a different geometric configuration, than the original case B design, with higher aspect ratio and reduced wing area. For instance noting from Appendix II (b) and Appendix VI, the lift-drag polar for Case B indicates at a lift coefficient of 1.6 the drag coefficient is 0.138. For the optimum case, it has been reduced to 0.12. This is a substantial drag reduction, allowing the aircraft to achieve a much higher cruise Mach number, at lower weight, lower cost and lower fuel burn. The climb rate in this configuration is 1600 ft/min. The aircraft price is lower also, at a mere 7.07 million dollars. In terms of size the optimum aircraft is similar to the CJ1+, but from a performance point of view closer to the CJ4. The optimiser performance is a function of the range of free variables allowed, starting values of these free variables, the limits on constraints etc. At an undergraduate level, students may not be able to appreciate the full intricacies of the optimiser operation, so caution needs to be exercised in its use. The feature can be used to generate an optimum around the baseline design, by letting the software set the allowable range to be a percentage around the baseline value automatically. The constraints are then selected from a predefined list, and then selecting what needs to be minimised. In this case MTOW was chosen.

Key Parameters	Case B	Optimum
MTOW (kg)	6116	5700
EW (kg)	4602	4481
Aspect Ratio	8	10.3
Cruise Mach No.	0.7M	0.8M
BFL (m)	988	980
Thrust/Engine (lbs)	2100	1990
Wing Area (m^2)	27.7	23.6
Stall Speed (m/s)	13.75	12.98
DOC ($)	5439	5024

Table 6-22: Comparison of the Case (B) design and optimum design

Availability of software such as iADS means, that students are able to experiment with changing the parameters and seeing the results instantly. iADS is not meant to replace the conventional long hand calculations, instead it is intended as a guide, so that the students have a reasonably accurate estimate of the key design variables for them to aim at. It is meant to aid understanding.

By using the DATCOM interface an input file can easily be produced, such as in Appendix III (b) applicable to this case study. This allows students to use DATCOM software freely available in the public domain for dynamic stability analysis, should they require. Appendix IV (b) shows the MATLAB '.m' file of the designed aircraft produced by the software that contains the longitudinal and lateral stability derivatives computed using the analytical approach, along with other data enabling a comprehensive dynamic simulation to be produced, either in the MATLAB environment or in a flight simulator.

6.3 Conclusions

Two case studies were presented in this chapter, emulating the typical design procedure an undergraduate would follow in the course of aircraft design. The iterative nature of the process was highlighted. It was also indicated, that meeting all the design objectives in one or few iterations is difficult, and if it were to be the case, then prior knowledge of aircraft design and the variable sensitivities must be known. Even though the design looks viable, it is no way certain to be dynamically stable. This must be done by computing the stability and control derivatives, and the conventional stability equations need to be formulated, and their roots found. Incorporation of dynamic stability in the preliminary phase allows the students to gain confidence in their design before the design is verified in a flight simulator.

Chapter SEVEN

Conclusions and Recommendations

7.1 Conclusions

The long journey of creation and innovation in aviation history shows that aircraft design is not simple. It is truly a multidisciplinary subject. It hinges on aerodynamics, structures, propulsion, and stability and control to name a few. Modern aircraft are equipped with many advanced electronic and software systems too. The aircraft design process consists of three phases which are conceptual, preliminary, and detailed. Many universities in the UK, follow Problem-Based Learning approach in teaching aircraft design. Since, the aircraft design process is iterative in nature, and indeed non-unique, it can be said that many aspects of learning that are required, cannot be fully explored in the duration of the course of study. And it is fair to say, that some aspects of learning are likely to be shallow, due to limited exposure and time. However, at the end of the project, students gain valuable insight into the design process, and this prepares them well to embark upon a career in the aircraft industry. Most universities present the preliminary design project as in course assessment, and pertinent feedback is given on a weekly basis, to steer the students towards a feasible design.

The iterative nature of aircraft design lends itself to automation by coding the appropriate design methodology in a computer application. Most aerospace undergraduate students do not possess the capability to produce computer applications, which is the domain of computer scientists. However, students today are well versed with spreadsheets, and will produce relevant computational workbooks, that facilitate the aircraft design process. These workbooks are created incrementally and evolve over the entire design process. Since, the students learn on a need to know basis, part of the Problem-Based Learning approach, they are unlikely to do the full design iteration. The integration of computer software into Problem-Based Learning can encourage students to learn better, deeper, and in a much more interesting environment than before, by varying the ways in which information is delivered. SYNAC II and CAAD programs were the early aircraft design software which were developed in the mid of sixties and the beginning of the seventies. Thereafter, several software packages have been released into the market such as PIANO, AAA, and RDS. Most of these packages have been designed for use by aircraft design professionals. Also, their main purpose is the preliminary aircraft design in a commercial environment, and they are not intended for instructional use.

Aircraft design is not trivial; many design variables have to be manipulated and some variables have a pronounced effect on the outcome. Some of them are selected, while others are evaluated. As the design complexity increases, so do the number of variables that are used to represent complexity. In the preliminary design phase, the input variables are grouped according to aircraft components and mission requirements. All equations required for analysing this phase are also grouped into sections for clarity and easy understanding.

Based on the mathematical analysis of the preliminary design phase, a software package for undergraduate teaching was developed, and presented as part of this research. Torenbeek, Raymer, and Nelson methods and equations, which are widely used, are employed for design synthesis and for aspects of dynamic stability. Due to the fact that the GUI is as important as the software computations itself, an OOUI is implemented to dress up the software functions. The synthesis program does everything needed in the preliminary design environment. This includes computation of aircraft geometry, weights, CG and aerodynamics, aspects of flight performance, cost estimation, numerical optimisation and dynamic stability. The output data are displayed in text form, as well as it can be saved as a spreadsheet for further analysis. A parametric studies module is implemented to obtain the majority of the design parameters and its sensitivity. Although, Torenbeek's equation for estimating BFL is implemented in the synthesis program, an additional module is employed to synthesise the takeoff stage in a more accurate analysis. It is essential for students to evaluate both static and dynamic stability concepts early in the preliminary design phase. The dynamic stability module is implemented based on the analytical approach of the aircraft equations of motion. This early evaluation, before the detailed design phase takes place, makes the design process more efficient and speeds up the design process itself. 3-View of the proposed aircraft is available, as well, and changes dynamically with the design variable variations to explore the influence of these variables graphically on the design.

In the preliminary design phase of the interactive design process, where the maximum takeoff weight is broken down into its components and sub-components, a modified module is developed to increase the prediction accuracy to better than 4%. Its output results agree very favourably with the published data of current Airbus and Boeing aircraft.

Estimating the DOC and SMC are important from a commercial viability of designs. Three methodologies exist, formulated by the AEA, ATA and NASA. Each method for determining the DOC and SMC differs but produces comparable results. This work summarises all the three methodologies, and will serve as a source for cost estimation for future aircraft design engineering students and professionals alike. They have been applied to predicting the cost of existing transport aircraft and the results show that ATA and NASA methodologies are close to each other and agree with the latest available data of transport aircraft costs.

A typical design process due to its iterative nature is time consuming, and a feasible design is arrived at, as an accumulation of knowledge and experience. Once a feasible design is identified, it should be refined. This is done by optimisation. The constrained optimisation program RQPMIN developed by the RAE, was re-coded using Pascal for use in iADS. This allows one of five objective functions to be minimised subject to equality or inequality constraints, and the number of free variables that could influence the design. Only after a few optimised preliminary designs are complete, the final stage of the detailed analysis can be done, from which a candidate for prototyping is normally chosen. The iADS environment is efficient, and all the key and relevant design variables are available to the user for manipulation.

From an undergraduate point of view, the logical grouping of design variables and a graphical interface facilitates a better understanding of the subject; this can be done in many ways as a 3-view graphic or via the parametric studies, as a one-to-one or two-to-one variation. This is a key feature that enables a robust picture of parameter variation to be produced, and the associated sensitivities.

The availability of a software such as iADS means, that students are able to experiment with changing the parameters and seeing the results instantly. The iADS is not meant to replace the conventional long-hand calculations, instead, it is intended as a guide, so that the students have a reasonably accurate estimate of the key design variables for them to aim at. It is meant to aid understanding. The comparisons of **Table 6-11** and **Table 6-21** reveal that the designed aircraft are similar to the Boeing 737-700 and Cessna CJ4, respectively. And it can therefore be concluded, that the iADS software produces satisfactory results that can form the basis for the detailed design phase.

The iADS software has some assumptions and limitations, since it is based mainly on semi-empirical methods. So, first of all, the software implements methodologies limited to the design of conventional jet transport aircraft. Secondly, the centres of mass for systems and surface control are assumed to be in the mid-point of the cabin section, mainly due to the absence of detailed data available. Other limitation is that all fuel tanks are located in the wing. There is no provision for locating fuel tanks in the fuselage or the tail section of the aircraft. In aerodynamics module, the lift slope of the horizontal tail is assumed to be the same as the 2D slope of the wing. Also, in calculating the maximum lift coefficient of the trimmed aircraft, the effects of the fuselage and the nacelle lift are very small and are ignored. Hence, the maximum lift coefficient of the aircraft minus tail is assumed to be the same as the wing alone. Finally, the ambiguous definition of the range in NASA cost estimation methodology as small, medium and long is replaced by the values <2000, 2000- 4000, and >4000 nautical mile, respectively. These values seem reasonable in calculating the utility.

The dynamic stability module implements stability equations that are based on small perturbations, and in the linear region. Non linear effects are not considered. This is mainly because, that in preliminary design phase detailed calculations are not required. The sole intention of this phase is to arrive at one or two candidate designs, on which detailed calculations could be performed. Therefore, additional complexity due to cross-coupling between variables and non-linearities are excluded to keep the computations simple. Neither are the environmental conditions, such as gusts considered here.

A new method for estimating the MTOW has been proposed. To facilitate this work, the output from this research is a software package for aiding the understanding of students for whom designing an aircraft is a new experience. This is a long process and the student would have to have spends a larger number of hours programming all the equations in as well as setting these inside a design framework and sorting the optimisation and display functions. This process is iterative with no guarantee that design solutions can be found. The consideration of the learner and his/her attitude to the activity is a valuable contribution and something that can continue to develop in the future. The contribution from this research to the scientific literature will mature more when papers can be published on the impact of the learning environment on the students and on how the staff should use the package to create a positive and enhanced learning experience.

7.2 Recommendations

- The iADS is a dedicated development for subsonic civil transport aircraft, with MTOW in excess of 5 tonnes approximately. It should be extended to design general aviation aircraft.

- For a comprehensive design and analysis feature, the software should be modified to produce the necessary mesh for use in a CFD package such as Phoenics [141] or STAR-CCM+ [142]. This will allow aerodynamic simulations to be produced.

- The DATCOM software at present is stand alone. It should be incorporated into the iADS software for ease of use.

- A feature must be incorporated that would allow any formula or calculation performed by the program to be substituted by the user supplied methodology or function, a Pascal compiler must therefore be embedded into iADS.

- A generic turbo-fan engine is in-built in iADS, the software should be extended to cater for turbo-prop and piston engine aircraft.

- A database of all existing aircraft should be created as part of the iADS software permitting the user to pick out aircraft having similar configuration and performance to the user specified input information. This can be done by the user selecting a range of parameters, which are then matched to the database, and rapid baseline values for other parameters are then easily determined, further reducing the preliminary design time.

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Appendix I

Input Data File Description for DATCOM Software

The for005 file (aka input file) defines the flight characteristics and geometry of the aircraft. It follows a very specific format and any deviance for the format will cause the analysis not to be run. First, input the CASEID on the first line:

CASEID ---------- MY DESIGN -----------------

Every piece of data that defines the flight characteristics and geometry of the aircraft is contained in name lists. In the input file, name lists are preceded by a "$" sign (ex. $FLTCON) and the statements are terminated with "$" signs. The followings are the name lists:

1. **FLTCON -** defines the flight conditions
2. **SYNTHS -** locates the cg, wing, horizontal tail, and vertical tail with respect to a reference line
3. **BODY -** defines the body geometry
4. **WGPLNF -** defines the wing geometry
5. **HTPLNF -** defines the horizontal tail geometry
6. **VTPLNF -** defines the vertical tail geometry

1- FLTCON

The FLTCON name list defines the flight conditions such as Mach number(s), altitude(s), and angle of attacks to be analysed. A full list of the inputs can be found on page 27 of the Digital DATCOM manual volume I [110]. The following variables are used for the FLTCON name list:

1. **NMACH -** number of Mach numbers to be run. For example, only one Mach.
2. **MACH -** the mentioned Mach numbers to be run. For example, at Mach = 0.60.
3. **NALPHA -** the number of angles of attack to test. For example, 10 of AOA's.
4. **ALSCHD -** the schedule of angles of attack. For example, (in deg): -4, -2, 0, 2, 4, 6, 8, 10, 12, 14 (ten total)
5. **NALT -** number of altitudes to run. For example, only one altitude.
6. **ALT -** the altitudes to run. For example, at 5000 feet.
7. **WT -** the weight of the aircraft.
8. **LOOP** - set to 0, 1, or 2. The value '0' for LOOP parameter, enables the user to run cases at fixed altitude with varying Mach number; '1' for fixed Mach number with varying altitudes; and '2' for varying Mach numbers and altitudes together.

Note that each number must contain a decimal point. Also, the file can only be a certain number of characters in horizontal length, so do not go over. The order, in which the variables are given, are not crucial, but the following format is recommended. The following example describes what goes into the input file:

$FLTCON NMACH=1.0,MACH(1)=0.6,NALPHA=10.0,ALSCHD(1)=-4.0,-2.0,
0.0,2.0,4.0,6.0,8.0,10.0,12.0,14.0,NALT=1.0,ALT(1)=5000.0, WT=13395.0,LOOP=1.$

2- SYNTHS

The SYNTHS (Synthesis) name list is very important because it sets up the CG location as well as the position of the wing and tail surfaces. Page 33 in DATCOM volume I [60] indicates exactly what dimensions are needed.

All horizontal measurements are taken from the nose of the aircraft. All vertical measurements are taken from a reference line conveniently placed at the centre of the aircraft. The following variables are used for the SYNTHS name list:

1. **XCG -** the horizontal position of the CG.
2. **ZCG -** the vertical position of the CG with respect to the reference line.
3. **XW -** the horizontal position of the apex of the wing. For example, 3.63 feet.
4. **ZW -** the vertical position of the wing apex with respect to the reference line. For example, 0.42 feet.
5. **ALIW -** the incidence of the wing in degrees. For example, 1 degree.
6. **XH -** the horizontal position of the apex of the horizontal tail. For example 28.73 feet.
7. **ZH -** the vertical position of the horizontal tail apex with respect to the reference line. For example, 5.24 feet.
8. **ALIH -** incidence of the horizontal tail. For example, 0 degrees.
9. **XV -** the horizontal position of the apex of the vertical tail. For example, 18.3 feet.
10. **ZV -** the vertical position of the vertical tail apex with respect to the reference line. For example, 0 feet.

The following example describes what goes into the input file:

$SYNTHS XCG=11.17,ZCG=0.0,XW=3.63,ZW=0.42,ALIW=1.0,XH=28.73,
ZH=5.24,ALIH=0.0,XV=18.3,ZV=0.0$

3- BODY

The variables of the name list BODY are:

1. **NX -** number of body stations
2. **X -** horizontal distance of each station
3. **S -** cross sectional area at each station

The following example describes what goes into the input file:

$BODY NX=8.0, X(1)=0.0,0.74,8.35,13.14,19.35,24.41,28.41,30.77,
S(1)=5.19,9.32,16.89,16.89,15.94,11.12,5.85,2.5$

4- WGPLNF

WGPLNF name list is used to define the wing geometry. DATUM volume I page 37 and 38 can be extremely helpful for all of the plane forms. The following variables are used in defining the wing:

1. **CHRDTP -** the length of the chord at the tip of the wing. For example, 7.02 feet.
2. **SSPNOP** – it is the "Semi-Span outboard panel". For example, 11.32 feet.
3. **SSPNE -** the "exposed" semi-span is measured from the side of the fuselage to the tip chord. For example, 13.41 feet..
4. **SSPN -** the theoretical semi-span, which is b/2. This dimension is from the root chord to the tip chord. For example, 15.71 feet.
5. **CHRDBP -** the chord at the break point between the inboard and outboard panel. For example, 8.4 feet.
6. **CHRDR -** the length of the chord at the root of the wing. For example, 14.0 feet.
7. **SAVSI -** the sweep of the wing at the inboard panel. For example, 45 degrees.
8. **SAVSO -** the sweep of the wing at the outboard panel. For example, 45 deg.
9. **TWISTA -** the twist angle of the wing (wash-out). For example, 0 deg.
10. **CHSTAT -** the % of the *mac* at which the sweep angle will be referenced. Usually this is c/4 or 0.25.
11. **DHDAHI -** the dihedral of the inboard panel. If the inboard and outboard panel dihedral is the same (constant dihedral across the wing), then only DHDADI is inputted. For example, the wing dihedral is -3 deg.
12. **TYPE -** different plane form types (refer to manual). Will be set to 1.

These values are inputted into the for005 file as follows:

$WGPLNF CHRDTP=7.02,SSPNOP=11.32,SSPNE=13.41,SSPN=15.71,
CHRDBP=8.4,CHRDR=14.0,SAVSI=45.0,SAVSO=45.0,CHSTAT=0.25,
TWISTA=0.0,DHDADI=-3.0,TYPE=1.0$

5- HTPLNF

The horizontal tail is described the same way as wing using the same variables. Because the geometry of the HT is simpler, the following variables are used: **CHRDTP**, **SSPNE**, **SSPN**, **CHRDR**, **SAVSI**, **CHSTAT**, and **TYPE**.

For example:

1. **CHRDTP** is 1.86 feet
2. **SSPNE** is 5.42 feet
3. **SSPN** is 5.43 feet
4. **CHRDR** is 4.69 feet
5. **SAVSI** is 45 degrees
6. **CHSTAT** is 0.25
7. **TYPE** is 1

And so the values are inputted into the HTPLNF name list as follows:

$HTPLNF CHRDTP=1.86,SSPNE=5.42,SSPN=5.43,CHRDR=4.69,SAVSI=45.0,
CHSTAT=0.25,TYPE=1.0$

6- VTPLNF

The vertical tail is described the same way as wing using the same variables. Because the geometry of the VT is simpler, the following variables are used: **CHRDTP**, **SSPNE**, **SSPN**, **CHRDR**, **SAVSI**, **CHSTAT**, and **TYPE**.

For example:

1. **CHRDTP** is 3.76 feet
2. **SSPNE** is 6.05 feet
3. **SSPN** is 8.18 feet
4. **CHRDR** is 12.47 feet
5. **SAVSI** is 55 degrees
6. **CHSTAT** is 0.25
7. **TYPE** is 1

And so the values are inputted into the VTPLNF name list as follows:

$VTPLNF CHRDTP=3.76,SSPNE=6.05,SSPN=8.18,CHRDR=12.47,SAVSI=55.0,
CHSTAT=0.25,TYPE=1.0$

AEROFOIL DESIGNATIONS

Page 74 and 75 of volume I has some information on this. If the wing was a NACA-2412, then input into the for005 file: NACA-W-4-2412. The "NACA" indicates that it is a NACA aerofoil, for which DATCOM has built in data. "W" indicates that the aerofoil is to be applied to the wing ("H" for HT, and "V" for VT). "4" indicates that it is a 4 series aerofoil (DATCOM accepts 1, 4, 5, and 6 series aerofoils). And after that is the aerofoil designation number, in this example it is "2412."

For the NACA 66-012 aerofoil is used for the wing, and the 66-009 for the tails. They will simply be inputted into DATCOM as follows:

NACA-W-6-66-012 NACA-H-6-66-009 NACA-V-6-66-009

TERMINATING THE FILE

There are a number of commands that can be placed at the end of the file to specify information and give more data. For example, the following commands are used:

1. **DIM FT -** specifies that all of the dimension were given in feet and all out the output should be in "English" units
2. **BUILD -** show the data for all of the components. not just the aircraft
3. **PLOT -** generate a plot file (for013.dat) to input into the MATLAB plotting program

Finally, the file ends with "NEXT CASE". In the input file this looks like:

DIM FT
BUILD
PLOT
NEXT CASE

Appendix II (a)

iADS Text Output for Case Study (A)

```
==============================================================
                     Wing      Tailplane     Fin
Aspect Ratio          9.00        5.00       1.80
Gross Area    (m^2) 122.00       21.33      15.22
Span            (m)  33.14       10.33       5.23
Taper Ratio           0.27        0.30       0.30
Thickness chord ratio 0.13        0.12       0.13
MeanAeroChord   (m)   4.09        2.07       2.91
Chord at C.line (m)   5.80        3.18       4.47
Tail Arm        (m)              18.71      18.59
Sweep Angle  (deg.)  25.00                  40.00

Wing Location 12.29 (m) from nosecone apex to the leading edge at
centreline
==============================================================
                    Fuselage    One Nacelle
Diameter       (m)    3.80        1.75
Total Length   (m)   36.15        3.50
NoseConeLength (m)    4.00        2.00
Cabin Length   (m)   20.75
TailConeLength (m)   11.40        1.50
Wetted Area  (m^2)  385.60       14.95

Number of Passengers           150.00
Total Take-off Thrust (lbs)   45539.90
Engine scale factor              3.35

Load Gust Factor                 5.885
Manoeuvre Load Factor            4.125
Design Dive Speed (I.A.S.)(m/s) 304.000

========== Mass calculation All Mass in Kg ===================
Wing includes flaps           6653.2
Fuselage                      7958.0
Empennage                     1010.8
Nacelles                      1136.1
Engines                       4054.2
Propulsion System             5584.1
Propulsion (total)            6720.2
Undercarriage                 2650.2
Surface Controls              1077.5

Auxiliary power unit           143.9
Paint & Oxygen system          602.4
Electrical system             1420.3
Avionics, Instruments, AP      962.2
Air conditioning & Anti-icing  679.1
Hydraulic system               762.5
Systems (Total)               4570.3

Furnishings                   3660.6
Empty Mass                   34300.8

Operation Items               1292.5
Crew mass                      186.0
```

```
Flight attendants                340.0
Op. empty mass                 36119.4

Passenger Load                 18030.5
Zero Fuel mass                 54149.9
Total Fuel                     16000.0

Maximum Take-Off (MTOW)        70149.9

============================================================

C.G. Position From Nose Apex.
Empty aircraft (m)                16.74
Datum 50%fuel full payload (m)    15.98
Aft Limit                         16.99
Forward Limit(m)                  15.92

===========    Aerodynamics Data ==============================
== Zero-Lift Drag Coefficients ==
Exposed Wing                      0.006435
Fuselage                          0.007711
Nacelles(total)                   0.009710
Horizontal Tail                   0.001489
Vertical Tail                     0.001032
Interference                      0.000584

      CL          CD
     ----      --------
     0.00      0.027647
     0.20      0.028784
     0.40      0.033436
     0.60      0.041603
     0.80      0.053284
     1.00      0.068479
     1.20      0.087189
     1.40      0.109414
     1.60      0.135153

===========    Static Stability ==============================
Neutral Point
      POWER-OFF (m)             17.34 from Nose
Static Margin
      DATUM C.G., POWER-OFF     0.33

===========    Max Lift Requirement ==========================
         Flap Defln.  Section CL   Wing CL  Trimmed a/c CL
            (deg)       (max)       (max)       (max)
Takeoff     23.00        3.16        2.32        2.21
Landing     30.00        3.46        2.52        2.39

===========    Cruise Mach Number ============================
Cruise mach no. =   0.84

===========    MISSION STAGE ANALYSIS ==========================
== first stage ==
Initial mass (kg)         70149.9
                   climb       cruise      descent
Distance   (m)  160963.5    4895000.0     134280.4
Fuel burn (kg)    1051.0      12426.7          0.0
Time       (s)     983.5      19272.7        752.1
IAS      (m/s)     119.6        150.2        139.2
```

```
Cruise Altitude       (m) 9690.00
Cruise Thrust Setting (%)   86.00
Start of climb      (m/s)   14.43
End of climb        (m/s)    5.89
Start of descent    (m/s)  -14.80
End of descent      (m/s)  -11.19

== Diversion stage ==
Initial mass (kg)        56626.8
                    climb      cruise      descent
Distance   (m)    48710.0     49950.0      80621.9
Fuel burn (kg)      431.2       144.3          0.0
Time        (s)     344.2       389.7        497.5
IAS      (m/s)      119.6        93.5        139.2

Cruise Altitude       (m)    6096.00
Cruise Thrust Setting (%)      38.00
Start of descent    (m/s)     -13.46
End of descent      (m/s)     -11.16

============= Summary of fuel total fuel burn ===============
Total mission fuel    (kg)  13523.0
Inc. ground man. fuel (kg)     45.0
Diversion fuel burn   (kg)    575.0
Holding fuel burn     (kg)   1000.0
Ground total fuel burn(kg)  15098.0
Average stage time     (s)  21008.0

============= Field Performance =============================
Second segment gradient         0.007
Balanced field length (m)    2421.0
Takeoff stall speed (m/s)      64.5
Landing mass         (kg)   56627.0
Landing field length  (m)    1208.0
Landing stall speed (m/s)      55.8

============= WAT Performance ===============================
At ISA + 20 deg.    elevation   0
Second segment climb gradient   0.007


======== Cost Estimation (Dollars 2010) =====================

Stage length      (km)            5500.0
Fuel price        ($/USG)            2.15
Fuel used         (kg)           13523.0
Block time        (hours)            5.95
Price of airframe ($M)              58.69
Price of engines  ($M)               7.12
Price of aircraft ($M)              65.80

====== Cost/Flight (ATA method) in US Dollars ==============
Utilisation          (hours/year)  4414
Depreciation cost                 $8702.3
Insurance cost                     $212.1
Interest cost                     $7309.9
Standing charge (Dep+Ins)         $8914.4
Standing charge (Dep+Ins+Int)    $16224.3
Total Labour maintenance          $1771.5
Total Material maintenance        $2637.4
Aircraft maintenance              $7597.5
```

```
Fuel & oil cost                     $9681.4
Flight Crew cost                    $4351.6
Indirect cost                      $14905.9

Total Operating Costs/Flight       $52760.7

===================== DOC, SMC ===========================
                         Without Interest  With Interest
Total DOC/flight   ($)          30544.800      37854.800
Total DOC/mile  ($/nm)             10.300         12.700
Seat mile cost  (c/nm)              6.856          8.497
```

Appendix II (b)

iADS Text Output for Case Study (B)

```
==============================================================
                      Wing      Tailplane     Fin
Aspect Ratio          8.00        5.00        2.50
Gross Area    (m^2)  27.87        4.71        3.27
Span            (m)  14.93        4.86        2.86
Taper Ratio           0.30        0.30        0.30
Thickness chord ratio 0.12        0.12        0.12
MeanAeroChord   (m)   2.05        0.97        1.14
Chord at C.line (m)   2.87        1.49        1.76
Tail Arm        (m)               7.26        6.37
Sweep Angle  (deg.)  15.00                   30.00

Wing Location  6.05 (m) from nosecone apex to the leading edge at
centreline
==============================================================

                    Fuselage    One Nacelle
Diameter       (m)    1.50         0.75
Total Length   (m)   13.75         2.50
NoseConeLength (m)    4.00         2.00
Cabin Length   (m)    5.25
TailConeLength (m)    4.50         0.50
Wetted Area  (m^2)   54.78         4.50

Number of Passengers             6
Total Take-off Thrust (lbs)   4214.1
Engin scale factor               0.31

Load Gust Factor              12.710
Manouvre Load Factor             5.309
Design Dive Speed (I.A.S.) (m/s) 278

========== Mass calculation All Mass in Kg ===================
Wing includes flaps            757.9
Fuselage                      1143.6
Empennage                      118.9
Nacelles                       105.1
Engines                        438.2
Propulsion System              603.5
Propulsion (total)             708.7
Undercarriage                  325.5
Surface Controls               199.9

Auxiliary power unit            11.5
Paint & Oxygen system           68.6
Electrical system              212.7
Avionics & Instruments, +AP    199.2
Air cond & Anti-icing          116.9
Hydraulic system               311.2
Systems (Total)                920.0

Furnishings                    427.9
Empty Mass                    4602.4

Operation Items                 51.7
Crew mass                      186.0
```

```
Flight attendants                 0.0
Op. empty mass                 4840.1

Passsenger Load                 721.2
Zero Fuel mass                 5561.3
Total Fuel                      555.0

Maximum TakeOff                6116.3

============================================================
C.G. Position From Nose Apex.
Empty aircraft       (m)          7.73
Datum Position -
50%fuel full payload              7.57
Aft Limit            (m)          7.76
Forward Limit        (m)          7.52

===========    Aerodynamics Data ==============================
== Zero-Lift Drag Coefficients ==
Exposed Wing                      0.007359
Fuselage                          0.005608
Nacelles(total)                   0.004244
Horizontal Tail                   0.001660
Vertical Tail                     0.001120
Interference                      0.000653
=========================================================

      CL          CD
     ----       --------
     0.00      0.021533
     0.20      0.022685
     0.40      0.027669
     0.60      0.036484
     0.80      0.049132
     1.00      0.065612
     1.20      0.085924
     1.40      0.110068
     1.60      0.138044

===========    Static Stability ==============================

Neutral Point (POWER-OFF)         (m) =  7.88 from Nose
Static Margin (DATUM C.G., POWER-OFF) =  0.15

===========    Max Lift Requirement ==========================

         Flap Defln.  Section CL   Wing CL  Trimmed a/c CL
            (deg)        (max)       (max)       (max)
TakeOff =   22.00        3.11        2.44        2.39
Landing =   30.00        3.46        2.69        2.61

Cruise mach no. =   0.70

===========    MISSION STAGE ANALYSIS ==========================
== first stage ==
Initial mass (kg)              6116.3
 ----------------- climb ------ cruise ----- descent ---
Distance   (m) = 273029.8    2469750.0     108603.1
Fuel burn (kg) =     77.3        346.4          0.0
Time       (s) =   1604.8      11700.6        707.0
IAS      (m/s) =    119.57       124.84       120.00
```

```
Cruise Altitude       (m)       9690
Cruise Thrust Setting (%)         86.00

*********** R. O. D / C ************
Start of climb   (m/s)           12.65
End of climb     (m/s)            2.45
Start of descent (m/s)          -16.06
End of descent   (m/s)          -11.73


== Diversion stage ==
Initial mass (kg)               5647.2
----------------- climb ------ cruise ----- descent ---
Distance   (m) =  73651.8     49950.0      65438.8
Fuel burn (kg) =     30.4         6.8          0.0
Time       (s) =    516.5       389.7        468.7
IAS      (m/s) =    119.57       93.53       120.00

Cruise Altitude       (m)       6096
Cruise Thrust Setting (%)         38.00

*****************R. O. D / C ************
Start of descent (m/s)          -14.44
End of descent   (m/s) = -11.75

============= Summary of fuel total fuel burn ===============
Total mission fuel    (kg)       469
Inc. ground man. fuel (kg)        45
Diversion fuel burn   (kg)        37
Holding fuel burn     (kg)        47
Ground total fuel burn(kg)       554
Average stage time     (s)     14012

============= Field Performance ==============================

Second segment gradient   = 0.012
Balanced field length (m)        988
Takeoff stall speed (m/s)         38.3
Landing mass         (kg)       5647
Landing field length  (m)        614
Landing stall speed (m/s)         35.2

============= WAT Performance ================================
At ISA + 20 deg.elevation (m)      0
Second segment climb gradient      0.011


======== Cost Estimation (Dollars 2010) =====================

Stage length             (km)   2775
Fuel price             ($/USG)      2.15
Fuel used                (kg)    469
Block time             (hours)      4.01
Price of airframe        ($M)      6.00
Price of engines         ($M)      1.30
Price of aircraft        ($M)      7.30

====== Cost/Flight (ATA method) in US Dollars ==============
Utilisation          (hours/year) 4187
Depreciation cost        ($)     710.7
```

```
Insurance cost             ($)    17.0
Interest cost               ($)  597.0
Standing charge (Dep+Ins)   ($)  727.7
Standing charge (Dep+Ins+Int($) 1324.7
Total Labour maintenance    ($)  469.3
Total Material maintenance  ($)  266.7
Aircraft maintance          ($) 1580.6
Fuel & oil cost             ($)  342.7
Flight Crew cost            ($) 2788.9
Indirect cost               ($)  170.0

Total Operating Costs/Flight($) 6206.9

======================= DOC, SMC ============================
                          Without Interest  With Interest
Total DOC/flight   ($)          5439.9           6036.9
Total DOC/mile  ($/nm)             3.6              4.0
Seat mile cost  (c/nm)            60.504           67.144
```

Appendix III (a)

Input File of the Case Study (A) for DATCOM Software

```
CASEID --- My Design (iADS) ---
   $FLCON
    NMACH=1.0,
    MACH(1)=0.20,
    NALPHA=7.0,
    ALSCHD(1)=-4.0,-2.0, 0.0, 2.0, 4.0, 6.0, 8.0,
    NALT=1.0,
    ALT(1)=0.0,
    WT= 6 331.0,
    LOOP=1.0
   $

   $SYNTHS
     XCG= 0.00,
     ZCG= 0.00,
     XW=12.29,
     ZW= 0.00,
     ALIW= 2.00,
     XH=32.47,
     ZH= 1.30,
     ALIH=-2.00,
     XV=31.18,
     ZV= 1.9
   $

   $BODY
    NX=5.0,
    X(1)=  2.00,  8.00, 20.00, 32.00, 34.00,
    S(1)= 5.67, 11.34, 11.34, 11.34,  6.16,
   $

   NACA-5-64-413
   $WGPLNF
    CHRDTP= 1.57,
    SSPNOP=16.57,
    SSPNE=14.67,
    SSPN=16.57,
    CHRDBP= 5.22,
    CHRDR= 5.80,
    SAVSI=25.0,
    SAVSO=25.0,
    CHSTAT=0.00,
    TWISTA=0.3,
    DHDADI=0.0,
    TYPE=0.0
   $
```

```
NACA-4-0012
$HTPLNF
 CHRDTP= 0.95,
 SSPNE= 5.16,
 SSPN= 5.16,
 CHRDR= 3.18,
 SAVSI=25.0,
 CHSTAT= 0.25,
 TYPE1.0
$

NACA-4-0015
$VTPLNF
  CHRDTP= 1.34,
  SSPNE= 5.23,
  SSPN= 5.23,
  CHRDR= 4.47,
  SAVSI=40.0,
  CHSTAT= 0.25,
  TYPE1.0
$

DIM M
BUILD
PLOT
NEXT CASE
```

Appendix III (b)

Input File of the Case Study (B) for DATCOM Software

```
CASEID --- My Design (iADS) ---
   $FLCON
    NMACH=1.0,
    MACH(1)=0.2,
    NALPHA=7.0,
    ALSCHD(1)=-4.0,-2.0, 0.0, 2.0, 4.0, 6.0, 8.0,
    NALT=1.0,
    ALT(1)=0.0,
    WT=  5727.0,
    LOOP=1.0
   $

   $SYNTHS
     XCG= 7.57,
     ZCG= 0.00,
     XW= 6.05,
     ZW= 0.00,
     ALIW= 2.00,
     XH=11.76,
     ZH= 1.30,
     ALIH=-2.00,
     XV=11.49,
     ZV= 0.75
   $

   $BODY
     NX=5.0,
     X(1)=  2.00,  5.00,  7.00, 10.00, 12.00,
     S(1)=  0.93,  1.77,  1.77,  1.77,  0.84,
   $

   NACA-5-64-413
   $WGPLNF
     CHRDTP= 0.86,
     SSPNOP= 7.47,
     SSPNE= 6.72,
     SSPN= 7.47,
     CHRDBP= 2.58,
     CHRDR= 2.87,
     SAVSI=15,
     SAVSO=12.5,
     CHSTAT=0.00,
     TWISTA=0.3,
     DHDADI=0.0,
     TYPE=0.0
   $
```

```
NACA-4-0012
$HTPLNF
  CHRDTP= 0.45,
  SSPNE= 2.43,
  SSPN= 2.43,
  CHRDR= 1.49,
  SAVSI=15,
  CHSTAT= 0.25,
  TYPE1.0
$

NACA-4-0015
$VTPLNF
  CHRDTP= 0.53,
  SSPNE= 2.86,
  SSPN= 2.86,
  CHRDR= 1.76,
  SAVSI=30.0,
  CHSTAT= 0.25,
  TYPE1.0
$

DIM M
BUILD
PLOT
NEXT CASE
```

Appendix IV (a)

iADS MATLAB m' File for Case Study (A)

```
%  This file includes ALL variables required for STABILITY and CONTROL
analysis

%######################################################################
%  WING VARIABLES
W_AR          =      9.00;    %  Aspect Ratio
W_ARx         =      8.61;    %  Exposed Wing Aspect Ratio
W_TR          =      0.27;    %  Taper Ratio
W_TC          =      0.13;    %  Thickness Ratio
W_S           =    122.00;    %  Area (m^2)
W_Sx          =     99.97;    %  Wing Area (m^2)
W_B           =     33.14;    %  Span (m)
W_Rt          =      5.80;    %  Root (m)
W_swp         =     25.00;    %  SweepBack Angle (deg.)
W_inc         =      2.00;    %  Incidence Angle (deg.)
W_dh          =      2;       %  Dihedral Angle (deg.)
W_P           =      0.34;    %  Position to Fuselage Length Ratio
W_cla         =      6.10;    %  Aerofoil Lift Slop Curve
W_cm0         =     -0.08;    %  Aerofoil Pitching Moment Coefficient
W_alfa0       =     -4;       %  Aerofoil Alfa at Zero Lift (deg.)

%######################################################################
%  HORIZONTAL TAIL VARIABLES
HT_AR         =      5.00;    %  Aspect Ratio
HT_TR         =      0.30;    %  Taper Ratio
HT_TC         =      0.12;    %  Thickness Ratio
HT_V          =      0.80;    %  Volume Coeff
HT_S          =     21.33;    %  Area (m^2)
HT_B          =     10.33;    %  Span (m)
HT_Rt         =      3.18;    %  Root (m)
HT_Se         =      5.08;    %  Elevator Area (m^2)
HT_qh         =      0.95;    %  Pressure Ratio
HT_eff        =      0.9;     %  Efficiency
HT_inc        =     -3;       %  Incidence Angle (deg.)
HT_dh         =      2;       %  Wing Dihedral Angle (deg.)
HT_cla        =      6.25;    %  Aerofoil Lift Slop Curve
HT_cm0        =      0;       %  Horiz. Tail Aerofoil Pitch Moment Coeff

%##############################################################################
%  VERTIACAL TAIL VARIABLES
VT_AR         =      1.80;    %  Aspect Ratio
VT_TR         =      0.30;    %  Taper Ratio
VT_TC         =      0.13;    %  Thickness Ratio
VT_V          =      0.07;    %  Vol Coeff
VT_S          =     15.22;    %  Area (m^2)
VT_B          =      5.23;    %  Span (m)
VT_Rt         =      4.47;    %  Root (m)
VT_Sr         =      4.57;    %  Rudder Area (m^2)
VT_eff        =      1;       %  Vertical Tail Efficiency
VT_swp        =     40.00;    %  Vertical Tail SweepBack Angle (deg.)
VT_cla        =      6.25;    %  Vertical Tail Aerofoil Lift Slop Curve

%##############################################################################
%  FUSELAGE VARIABLES
F_D           =      3.80;    %  Diameter (m)
F_L           =     36.15;    %  FLength (m)
F_1           =      4.00;    %  Nose Length (m)
F_2           =     20.75;    %  Cabin Length (m)
F_3           =     11.40;    %  TailCone Length (m)
F_ln          =     12.29;    %  Wing Position at Fuselage centre line (m)
F_inc         =      0;       %  Incidence (deg.)
```

```
%############################################################################
%  NACELLE VARIABLES
Nac_D          =       1.75;    %  Diameter (m)
Nac_L          =       3.50;    %  Length (m)
Eng_N          =       2;       %  Number of Engines

%############################################################################
#####
% VARIABLES FROM RUNNING SINGLE PASS PROGRAM
W_C            =       4.09;    % W_C = Wing aerodynamic chord (m)
Xcg            =      15.98;    % Xcg = CG normal distance (m)
Xmac           =      16.44;    % Xmac = Xac power off (m)
Wto            =   60331.33;    % Aircraft Weight (kg)
Stab_Height    =       0.00;    % Aircraft Height (m)
Stab_roh       =       1.22;    % Roh value for the given height
Uo             =      68.00;    % Aircraft Speed (m/s)
q              =    2830.07;    % q = Air Pressure (kg/m^2)
CL0            =       1.70;
CD0            =       0.15;
HT_ac          =       9.16;    % Distance from HT-Aero centre to CG (m)
VT_ac          =      19.05;    % Distance from VT-Aero centre to CG (m)
Z_v            =       2.15;    % Z Distance from VT-aero centre to fuselage
centre line (m)
Z_w            =       1.42;    % Z Distance from Wing Aero centre to fuselage
centre line (m)
a_y2           =      15.74;    % Distance from centre line to the outer side
of the ailron (m)
a_y1           =      11.60;    % Distance from centre line to the inner side
of the ailron (m)

%############################################################################
##########% ###### LONGITUDINAL COEFFICIENTS #####
Cl_alfa        =       5.4226;
Cm_alfa        =      -2.6521;
Cm_zero        =       0.2260;
Cl_alfa_dot    =       0.0000;
Cl_q           =       0.0000;
Cm_alfa_dot    =      -9.5097;
Cz_alfa        =      -5.5719;
Cz_alfa_dot    =      -2.0282;
Cm_q           =     -28.5790;
Cz_q           =      -6.0952;
Cz_delta_e     =      -0.3129;
Cm_delta_e     =      -1.4320;
CT_u           =      -0.2987;
Cx_u           =      -0.2987;
Cz_u           =      -3.3292;
Cm_u           =       0.0800;
Cx_alfa        =       1.0774;

%##################################################################
%##### LATERAL COEFFICIENTS ########
Cy_beta        =      -0.7088;
Cn_beta        =       0.0222;
Cl_beta        =      -0.0974;
Cy_p           =       0.6220;
Cn_p           =      -0.2125;
Cl_p           =      -0.5867;
Cy_r           =       0.8149;
Cn_r           =      -0.1696;
Cl_r           =       0.0408;
Cl_delta_a     =       0.0537;
Cn_delta_a     =       0.0219;
Cy_delta_r     =       0.0736;
Cn_delta_r     =      -0.0413;
Cl_delta_r     =       0.0048;

%#####################################################################
```

```
% ####### DIMENSIONAL COEFFICIENTS ######
Xu              =      -0.0302;
Zu              =      -0.3005;
Mu              =       0.000584;
Xw              =       0.0572;
Zw              =      -0.4689;
Mw              =      -0.0194;
Zw_dot          =      -0.0051;
Mw_dot          =      -0.0021;
X_alfa          =       6.1656;
Z_alfa          =     -31.8872;
M_alfa          =      -1.3162;
Z_alfa_dot      =      -0.3488;
M_alfa_dot      =      -0.1418;
Z_q             =      -1.0483;
M_q             =      -0.4263;
Z_delta_e       =      -1.7907;
M_delta_e       =      -0.7107;


%############################################################################
% ####### LATERAL DIRECTIONAL DERIVATIVES ##########
Y_beta          =      -4.0564;
N_beta          =       0.0664;
L_beta          =      -1.0776;
Yp              =       0.8673;
Np              =      -0.1548;
Lp              =      -1.5815;
Yr              =       1.1363;
Nr              =      -0.1235;
Lr              =       0.1099;
L_delta_a       =       0.5939;
N_delta_a       =       0.0655;
Y_delta_r       =       0.4214;
N_delta_r       =      -0.1235;
L_delta_r       =       0.0528;


%############################################################################
% ######### MOMENTS ############
M_moment        =  318877.85;
N_moment        = -1939915.42;
L_moment        = -6712266.34;

%##########################################################
Along           = [Xu Xw 0 -9.81;Zu Zw Uo 0;Mu+Mw_dot*Zu Mw+Mw_dot*Zw
M_q+Mw_dot*Uo 0;0 0 1 0];
Blong           =[0 CT_u;Z_delta_e 0;M_delta_e+Mw_dot*Z_delta_e 0;0 0];
Alat            =[Y_beta/Uo Yp/Uo -(1-Yr/Uo) 9.81/Uo;L_beta Lp Lr 0;N_beta Np
Nr 0;0 1 0 0];
Blat            =[0 Y_delta_r/Uo;L_delta_a L_delta_r;N_delta_a N_delta_r;0 0];
C               =[1.0000 0.0000 0.0000 0.0000;0.0000 1.0000 0.0000
0.0000;0.0000 0.0000 1.0000 0.0000;0.0000 0.0000 0.0000 1.0000];
D               =[0.0000 0.0000;0.0000 0.0000;0.0000 0.0000;0.0000 0.0000];

%############################################################
% To evaluate eiginvalues for LONGITUDENAL A MATRIX
eig(Along);

% To evaluate eiginvalues for LATERAL A MATRIX
eig(Alat);

%#############################################################################
syslong         =ss(Along,Blong,C,D);
syslat          =ss(Alat,Blat,C,D);
```

Appendix III (b)

iADS MATLAB m' File for Case Study (B)

```
%  This file includes ALL variables required for STABILITY and CONTROL
analysis

%#########################################################################
%  WING VARIABLES
W_AR           =    8.00;        %  Aspect Ratio
W_ARx          =    7.66;        %  Exposed Wing Aspect Ratio
W_TR           =    0.30;        %  Taper Ratio
W_TC           =    0.12;        %  Thickness Ratio
W_S            =    27.87;       %  Wing Area (m^2)
W_Sx           =    23.56;       %  Exposed Wing Area (m^2)
W_B            =    14.93;       %  Span (m)
W_Rt           =    2.87;        %  Root (m)
W_swp          =    15.00;       %  SweepBack Angle (deg.)
W_inc          =    2.00;        %  Incidence Angle (deg.)
W_dh           =    2;           %  Dihedral Angle (deg.)
W_P            =    0.44;        %  Wing Position to Fuselage Length Ratio
W_cla          =    6.10;        %  Wing Aerofoil Lift Slop Curve
W_cm0          =    -0.08;       %  Aerofoil Pitching Moment Coefficient
W_alfa0        =    -4;          %  Wing Aerofoil Alfa at Zero Lift (deg.)

%##########################################################################
%  HORIZONTAL TAIL VARIABLES
HT_AR          =    5.00;        %  Aspect Ratio
HT_TR          =    0.30;        %  Taper Ratio
HT_TC          =    0.12;        %  Tail Thickness Ratio
HT_V           =    0.60;        %  Tail Volume
HT_S           =    4.71;        %  Tail Area (m^2)
HT_B           =    4.86;        %  Span (m)
HT_Rt          =    1.49;        %  Tail Root (m)
HT_Se          =    1.12;        %  Elevator Area (m^2)
HT_qh          =    0.95;        %  Pressure Ratio
HT_eff         =    0.9;         %  Tail Efficiency
HT_inc         =    -3;          %  Incidence Angle (deg.)
HT_dh          =    2;           %  Dihedral Angle (deg.)
HT_cla         =    6.25;        %  Aerofoil Lift Slop Curve
HT_cm0         =    0;           %  Aerofoil Pitching Moment Coefficient

%#########################################################################
%  VERTIACAL TAIL VARIABLES
VT_AR          =    2.50;        %  Aspect Ratio
VT_TR          =    0.30;        %  Taper Ratio
VT_TC          =    0.12;        %  Thickness Ratio
VT_V           =    0.05;        %  Tail Volume
VT_S           =    3.27;        %  Area (m^2)
VT_B           =    2.86;        %  Span (m)
VT_Rt          =    1.76;        %  Root (m)
VT_Sr          =    0.98;        %  Area (m^2)
VT_eff         =    1;           %  Efficiency
VT_swp         =    30.00;       %  SweepBack Angle (deg.)
VT_cla         =    6.25;        %  Aerofoil Lift Slop Curve

%##########################################################################
%  FUSELAGE VARIABLES
F_D            =    1.50;        %  Diameter (m)
F_L            =    13.75;       %  Length (m)
F_1            =    4.00;        %  Nose Length (m)
F_2            =    5.25;        %  Cabin Length (m)
F_3            =    4.50;        %  TailCone Length (m)
F_ln           =    6.05;        %  Wing Position at Fuselage centre line (m)
F_inc          =    0;           %  Incidence (deg.)
```

```
%##############################################################################
%  NACELLE VARIABLES
Nac_D          =   0.75;        %  Diameter (m)
Nac_L          =   2.50;        %  Length (m)
Eng_N          =   2;           %  Number of Engines

%##############################################################################
#######
% VARIABLES FROM RUNNING SINGLE PASS PROGRAM
W_C            =   2.05;        % Wing aerodynamic chord (m)
Xcg            =   7.57;        % CG normal distance (m)
Xmac           =   7.38;        % Xac power off (m)
Wto            =   5601.09;     % Aircraft Weight (kg)
Stab_Height    =   0.00;        % Aircraft Height (m)
Stab_roh       =   1.22;        % Roh value for the given height
Uo             =   68.00;       % Aircraft Speed (m/s)
q              =   2830.07;     % Air Pressure (kg/m^2)
CL0            =   1.70;
CD0            =   0.15;
HT_ac          =   7.07;        % Distance from HT-Aero centre to CG (m)
VT_ac          =   6.18;        % Distance from VT-Aero centre to CG (m)
Z_v            =   1.17;        % Z Distance from VT-aero centre to fuselage
centre line (m)
Z_w            =   0.56;        % Z Distance from Wing Aero centre to fuselage
centre line (m)
a_y2           =   7.09;        % Distance from centre line to the outer side
of the ailron (m)
a_y1           =   5.23;        % Distance from centre line to the inner side
of the ailron (m)

%##############################################################################
% ###### LONGITUDINAL COEFFICIENTS #####
Cl_alfa        =   5.4420;
Cm_alfa        =   -1.0522;
Cm_zero        =   0.2250;
Cl_alfa_dot    =   0.0000;
Cl_q           =   0.0000;
Cm_alfa_dot    =   -6.2655;
Cz_alfa        =   -5.5955;
Cz_alfa_dot    =   -1.8136;
Cm_q           =   -16.4165;
Cz_q           =   -4.7519;
Cz_delta_e     =   -0.3148;
Cm_delta_e     =   -1.1164;
CT_u           =   -0.3069;
Cx_u           =   -0.3069;
Cz_u           =   -3.3292;
Cm_u           =   0.0800;
Cx_alfa        =   1.0040;

%######################################################################
%##### LATERAL COEFFICIENTS ########
Cy_beta        =   -0.6594;
Cn_beta        =   0.1508;
Cl_beta        =   -0.0974;
Cy_p           =   0.3443;
Cn_p           =   -0.2125;
Cl_p           =   -0.5967;
Cy_r           =   0.5458;
Cn_r           =   -0.1122;
Cl_r           =   0.0401;
Cl_delta_a     =   0.0555;
Cn_delta_a     =   0.0227;
Cy_delta_r     =   0.0873;
Cn_delta_r     =   -0.0372;
Cl_delta_r     =   0.0069;

%######################################################################
```

```
% ####### DIMENSIONAL COEFFICIENTS ######
Xu              =   -0.0763;
Zu              =   -0.7393;
Mu              =   0.005541;
Xw              =   0.1408;
Zw              =   -1.1587;
Mw              =   -0.0729;
Zw_dot          =   -0.0057;
Mw_dot          =   -0.0065;
X_alfa          =   14.1376;
Z_alfa          =   -78.7947;
M_alfa          =   -4.9556;
Z_alfa_dot      =   -0.3844;
M_alfa_dot      =   -0.4441;
Z_q             =   -1.0071;
M_q             =   -1.1637;
Z_delta_e       =   -4.4325;
M_delta_e       =   -5.2580;

%######################################################################
% ####### LATERAL DIRECTIONAL DERIVATIVES ##########
Y_beta          =   -9.2855;
N_beta          =   3.1884;
L_beta          =   -6.3850;
Yp              =   0.5322;
Np              =   -0.4933;
Lp              =   -4.2946;
Yr              =   0.8438;
Nr              =   -0.2604;
Lr              =   0.2885;
L_delta_a       =   3.6396;
N_delta_a       =   0.4790;
Y_delta_r       =   1.2298;
N_delta_r       =   -0.7874;
L_delta_r       =   0.4496;

%######################################################################
% ######### MOMENTS ############
M_moment        =     36331.78;
N_moment        =   -132130.72;
L_moment        =   -702765.45;

%#######################################################
Along           = [Xu Xw 0 -9.81;Zu Zw Uo 0;Mu+Mw_dot*Zu Mw+Mw_dot*Zw
M_q+Mw_dot*Uo 0;0 0 1 0];
Blong           =[0 CT_u;Z_delta_e 0;M_delta_e+Mw_dot*Z_delta_e 0;0 0];
Alat            =[Y_beta/Uo Yp/Uo -(1-Yr/Uo) 9.81/Uo;L_beta Lp Lr 0;N_beta Np
Nr 0;0 1 0 0];
Blat            =[0 Y_delta_r/Uo;L_delta_a L_delta_r;N_delta_a N_delta_r;0 0];
C               =[1.0000 0.0000 0.0000 0.0000;0.0000 1.0000 0.0000
0.0000;0.0000 0.0000 1.0000 0.0000;0.0000 0.0000 0.0000 1.0000];
D               =[0.0000 0.0000;0.0000 0.0000;0.0000 0.0000;0.0000 0.0000];

%#########################################################
% To evaluate eiginvalues for LONGITUDENAL A MATRIX
 eig(Along);
% To evaluate eiginvalues for LATERAL A MATRIX
 eig(Alat);

%######################################################################
 syslong        =ss(Along,Blong,C,D);
 syslat         =ss(Alat,Blat,C,D);
```

Appendix V

Cessna Citation cj4: Specification and Description

CITATION CJ4

December 2012, Revision F

MANUFACTURER .... CESSNA AIRCRAFT COMPANY

MODEL 525C

1. GENERAL DESCRIPTION

The Cessna Citation CJ4 is a low-wing aircraft with retractable tricycle landing gear and a "T" tail. A pressurized cabin accommodates a crew of two and up to nine passengers (eight is standard). Two Williams International Co., LLC (Williams) FJ44-4A Full Authority Digital Engine Control (FADEC) controlled turbofan engines are pylon-mounted on the rear fuselage. Fuel stored in the wings offers generous range for missions typical of this class aircraft. Space for baggage is provided in the nose and tailcone with additional storage space available in the cabin.

Multiple structural load paths and system redundancies have been built into the aluminum airframe. Metal bonding techniques have been used in many areas for added strength and reduced weight. Certain parts with non-critical loads such as the nose radome and fairings are made of composite materials to save weight. The airframe design incorporates anti-corrosion applications and lightning protection.

Cessna offers a third-party training package for pilots and mechanics, and various manufacturers' warranties as described in this book. Cessna's worldwide network of authorized service facilities provides a complete source for all servicing needs.

1.1 Certification

The Model 525C is certified to the requirements of U.S. 14 CFR Part 23, Commuter Category, including day, night, VFR, IFR, and flight into known icing conditions. It will also be certified for single pilot operations for U.S. registered aircraft and for steep approach operations. The Citation CJ4 is compliant with all RVSM certification requirements. (Note: specific approval is required for operation within RVSM airspace; Cessna offers a no charge service to assist with this process.)

The Purchaser is responsible for obtaining aircraft operating approval from the relevant civil aviation authority. International certification requirements may include modifications and/or additional equipment; such costs are the responsibility of the Purchaser.

1.2 Approximate Dimensions

Overall Height	15 ft 4 in (4.67 m)
Overall Length	53 ft 4 in (16.26 m)
Overall Width	50 ft 10 in (15.49 m)
Wing	
Span (does not include tip lights)	50 ft 5 in (15.37 m)
Area	330.0 ft^2 (30.66 m^2)
Sweepback (at 25% chord)	12.5 degrees
Horizontal Tail	
Span (overall)	22 ft 1 in (6.73 m)
Area	79.6 ft^2 (7.40 m^2)
Sweepback (at 25% chord)	20 degrees
Vertical Tail	
Height	7 ft 5 in (2.26 m)
Area	59.3 ft^2 (5.51 m^2)
Sweepback (at 25% chord)	49 degrees
Cabin Interior	
Height (maximum over aisle)	57 in (1.45 m)
Width (trim to trim)	58 in (1.47 m)
Length (forward pressure bulkhead to aft pressure bulkhead)	22 ft 4 in (6.81 m)
Landing Gear	
Tread (main to main)	12 ft 4 in (3.76 m)
Wheelbase (nose to main)	21 ft 2 in (6.45 m)

3

1. GENERAL DESCRIPTION (Continued)

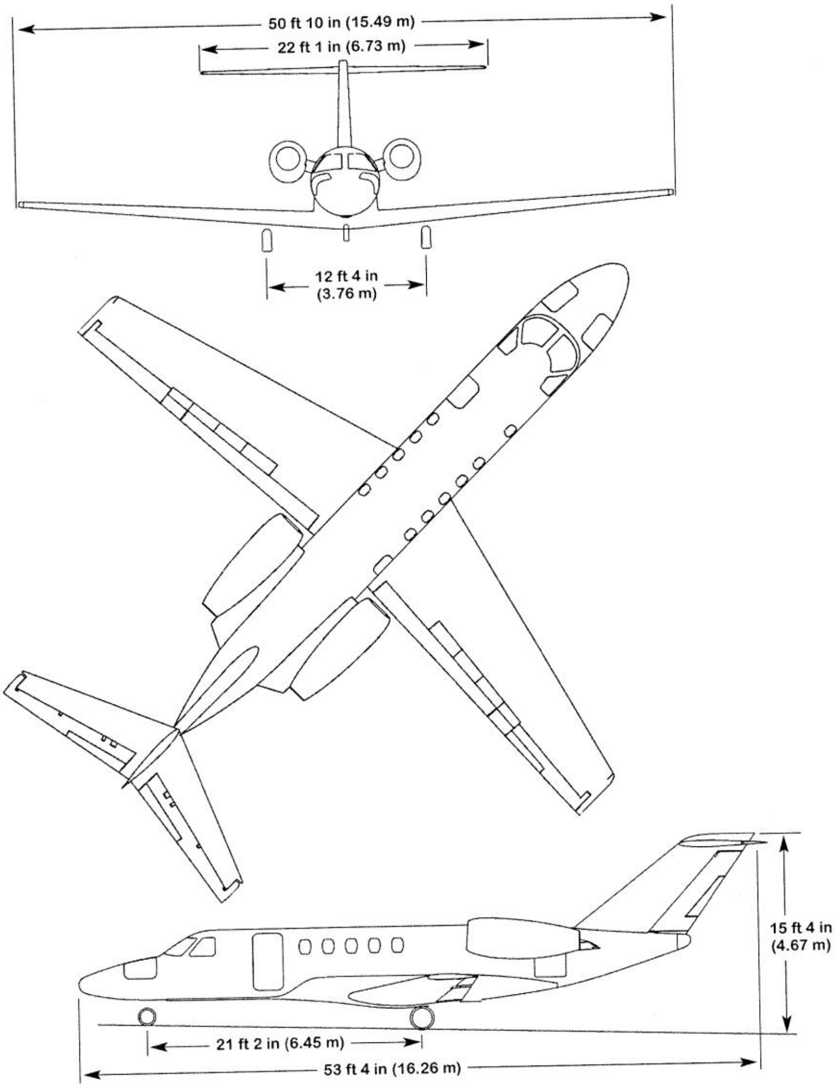


FIGURE I — CITATION CJ4 EXTERIOR DIMENSIONS

1. GENERAL DESCRIPTION (Continued)

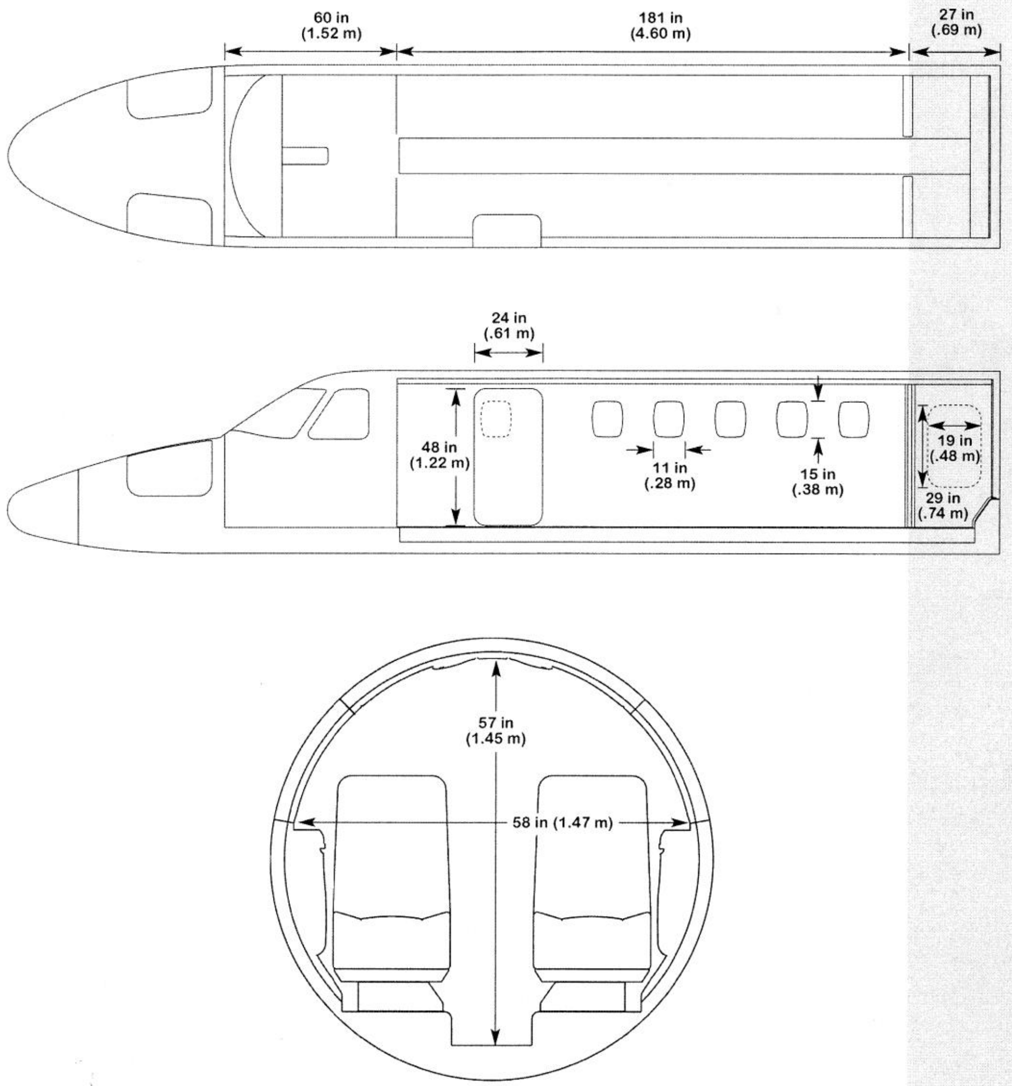


FIGURE II — CITATION CJ4 INTERIOR DIMENSIONS

1. GENERAL DESCRIPTION (Continued)

1.3 Design Weights and Capacities

Maximum Ramp Weight	17,230 lb (7,815 kg)
Maximum Takeoff Weight	17,110 lb (7,761 kg)
Maximum Landing Weight	15,660 lb (7,103 kg)
Maximum Zero Fuel Weight	12,500 lb (5,670 kg)
Standard Empty Weight*	9,920 lb (4,500 kg)
Full Fuel Payload	1,000 lb (454 kg)
Maximum Payload	2,180 lb (989 kg)
Fuel Capacity (usable) at 6.70 lb/gal	5,828 lb (2,644 kg)

* Standard empty weight includes unusable fuel, full oil, standard interior, and standard avionics.

2. PERFORMANCE

All performance data is based on a standard aircraft configuration, operating in International Standard Atmosphere (ISA) conditions with zero wind. Takeoff and landing field lengths are based on a level, hard surface, dry runway. Actual performance will vary with individual airplanes and other factors such as environmental conditions, aircraft configuration, and operational/ATC procedures.

Takeoff Field Length .... 3,190 ft (972 m)
(Maximum Takeoff Weight, Sea Level, 15° Flaps)

Climb Performance .... 29 min to 45,000 ft (13,716 m)
(Maximum Takeoff Weight, from Sea Level)

Maximum Altitude .... 45,000 ft (13,716 m)

Maximum Cruise Speed (± 3%) .... 451 KTAS (836 km/hr or 519 mph)
(14,000 lb, 31,000 ft (9,449 m))

NBAA IFR Range (100 nm alternate) (± 4%) .... 1,920 nm (3,556 km or 2,209 mi)
(Maximum Takeoff Weight, Full Fuel, Published Climb, High Speed Descent, and Maximum Cruise Thrust at 45,000 feet)

Landing Distance .... 2,740 ft (835 m)
(Maximum Landing Weight, Sea Level)

Appendix VI

iADS Output for Optimised Aircraft Design Case Study (B)

```
===============================================================
                    Wing      Tailplane     Fin
Aspect Ratio        10.38       5.00        2.50
Gross Area    (m^2) 23.61       3.02        2.81
Span            (m) 15.65       3.88        2.65
Taper Ratio          0.46       0.30        0.30
Thickness chord ratio 0.15      0.12        0.12
MeanAeroChord   (m)  1.58       0.78        1.06
Chord at C.line (m)  2.07       1.20        1.63
Tail Arm        (m) =====       7.40        6.57
Sweep Angle  (deg.) 15.00      =====       30.00

Wing Location  5.87 (m) from nosecone apex to the leading edge at
centreline
===============================================================
                  Fuselage    One Nacelle
Diameter       (m)    1.50        0.75
Total Length   (m)   13.75        2.50
NoseConeLength (m)    4.00        2.00
Cabin Length   (m)    5.25
TailConeLength (m)    3.15        0.50
Wetted Area  (m^2)   50.01        4.50

Number of Passengers              6.00
Total Take-off Thrust (lbs)    3981.30
Engin scale factor                0.29

Load Gust Factor                 10.622
Manoeuvre Load Factor             5.223
Design Dive Speed (I.A.S.)(m/s)  278.000

============= Mass calculation All Mass in Kg ==============
Wing includes flaps             666.9
Fuselage                       1063.4
Empennage                        92.6
Nacelles                         99.3
Engines                         415.7
Propulsion System               572.6
Propulsion (total)              671.9
Undercarriage                   346.2
Surface Controls                211.8

Auxiliary power unit             12.5
Paint & Oxygen system            71.4
Electrical system               212.7
Avionics, Instruments, & AP     209.1
Air cond. & Anti-icing          116.9
Hydraulic system                314.8
Systems (Total)                 937.4

Furnishings                     491.7
Empty Mass                     4481.8

Operation Items                  51.7
Crew mass                       186.0
```

```
Flight attendants              0.0
Op. empty mass              4719.5

Passenger Load              721.2
Zero Fuel mass              5440.7
Total Fuel                   259.3

Maximum Take-Off (MTOW)     5700.0


C.G. Position From Nose Apex.
Empty aircraft (m)             7.42
Datum 50%fuel full payload  (m) 7.30
Aft Limit            (m)       7.47
Forward Limit        (m)       7.24

==========   Aerodynamics Data   ========================
=== Zero-Lift Drag Coefficients ===
Exposed Wing                   0.008751
Fuselage                       0.006216
Nacelles(total)                0.005168
Horizontal Tail                0.001346
Vertical Tail                  0.001190
Interference                   0.000516

      CL          CD
     ----      --------
     0.00      0.024105
     0.20      0.024812
     0.40      0.028758
     0.60      0.035943
     0.80      0.046368
     1.00      0.060032
     1.20      0.076935
     1.40      0.097077
     1.60      0.120459

===========   Static Stability =========================

Neutral Point
       POWER-OFF (m)          7.55 from Nose
Static Margin
       DATUM C.G., POWER-OFF  0.16

===========   Max Lift Requirement ======================
         Flap Defln.  Section CL   Wing CL  Trimmed a/c CL
            (deg)        (max)       (max)       (max)
TakeOff     28.08        3.40        2.59        2.54
Landing     30.00        3.46        2.63        2.58

===========   Cruise Mach Number =========================
Cruise mach no. =   0.80

============   Mission Stage Analysis ====================
=== First stage ========
Initial mass (kg)        5700.0
                       climb       cruise      descent
Distance    (m)   153307.6    2423921.3     197771.2
Fuel burn (kg)        41.4        124.8          0.0
Time        (s)     1178.5      10252.9       1966.4
IAS      (m/s)       90.25       129.22        76.25
```

```
Cruise Altitude       (m)   10950.00
Cruise Thrust Setting (%)      51.60
Start of climb   (m/s)         14.07
End of climb     (m/s)          5.12
Start of descent (m/s)         -7.03
End of descent   (m/s)         -4.52

==== Diversion stage =====
Initial mass (kg)         5488.5
                    climb      cruise      descent
Distance   (m)    45415.2     34965.0     104619.8
Fuel burn (kg)       18.9         3.9          0.0
Time        (s)     428.2       420.8       1185.8
IAS      (m/s)       90.25       61.21        76.25

Cruise Altitude       (m)   5924.00
Cruise Thrust Setting (%)     26.60
Start of descent (m/s)        -5.64
End of descent   (m/s)        -4.46

=========== Summary of fuel total fuel burn ================
Total mission fuel    (kg)     212.0
Inc. ground man. fuel (kg)      45.0
Diversion fuel burn   (kg)      23.0
Holding fuel burn     (kg)      25.0
Ground total fuel burn(kg)     259.0
Average stage time     (s)   13398.0

=========== Field Performance =============================
Second segment gradient          0.025
Balanced field length (m)      980.0
Takeoff stall speed (m/s)       39.0
Landing mass         (kg)     5489.0
Landing field length  (m)      682.0
Landing stall speed (m/s)       38.0

============= WAT Performance ==============================
At ISA + 20 deg.elevation (m)     0
Second segment climb gradient    0.024

=========== Cost Estimation (Dollars 2010) =================

Stage length       (km)        2775.0
Fuel price      ($/USG)           2.15
Fuel used          (kg)         212.0
Block time      (hours)           3.84
Price of airframe  ($M)           5.80
Price of engines   ($M)           1.26
Price of aircraft  ($M)           7.07

====== Cost/Flight (ATA method) in US Dollars =============
Utilisation    (hours/year)    4161
Depreciation cost               $664.7
Insurance cost                   $15.9
Interest cost                   $558.3
Standing charge (Dep+Ins)       $680.6
Standing charge (Dep+Ins+Int)  $1238.9
Total Labour maintenance        $449.6
Total Material maintenance      $250.6
Aircraft maintance             $1509.5
```

```
Fuel & oil cost                  $158.2
Flight Crew cost                $2675.5
Indirect cost                    $161.1

Total Operating Costs/Flight   $5743.1

=================== DOC, SMC ================================
                        Without Interest   With Interest
Total DOC/flight   ($)          5023.7         5582.0
Total DOC/mile  ($/nm)             3.4            3.7
Seat mile cost  (c/nm)            55.875         62.084
============================================================
Convergence detected [code A] after 2065calls to user routine
```

Appendix VII

iADS pictorial user guide

iADS is a tool for preliminary aircraft design. At start-up the main interface screen is presented with top level menus these are the *File, Edit, Configure, Design, Output, Window, and Help*.

File, *Edit*, *Window*, and *Help* elements are general purpose menus, allowing opening, closing saving aircraft designs, since iADS supports MDI (multiple document interface) a *window* menu is provided to switch between opened documents. The software loads a default aeroplane with representative values, which can be altered using the *Configure* menu as in **Figure A-1**.

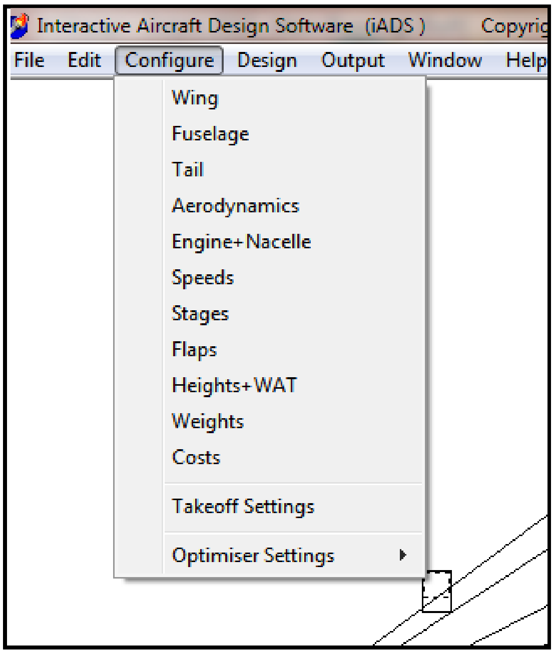
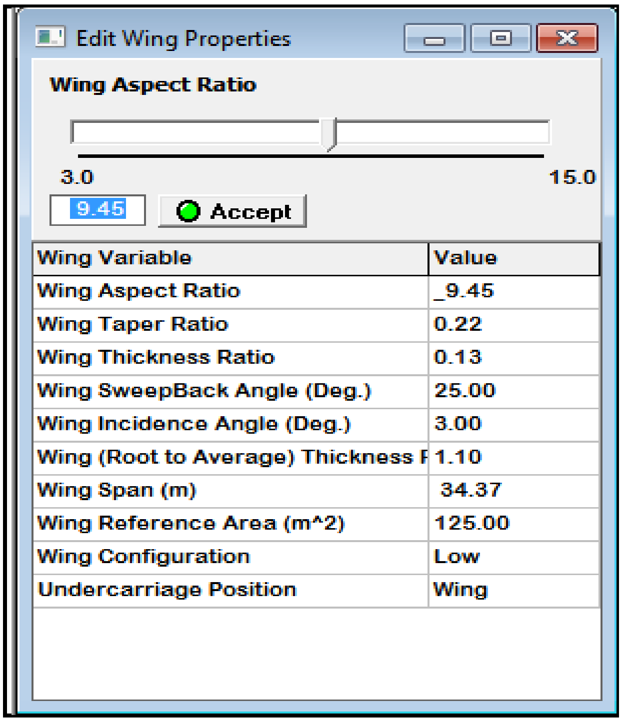


Figure A-1: Configure menu form

Each item in the menu has its own form which contains design variables. These forms are shown sequentially from **Figure A-2** to **Figure A-12.** In each form, the user can either enter his/her values directly in the 'Value' field or using the scroll bar. When sliding the scroll bar, its value is appeared in the text box below the scroll bar. Clicking '**Accept**' button, the text box value is saved to the associated design variable. The forms associated with the *Optimiser Settings* menu have already been presented in the main text **Figures 4-16, 4-17,** and **4-18**.



Edit Wing Properties

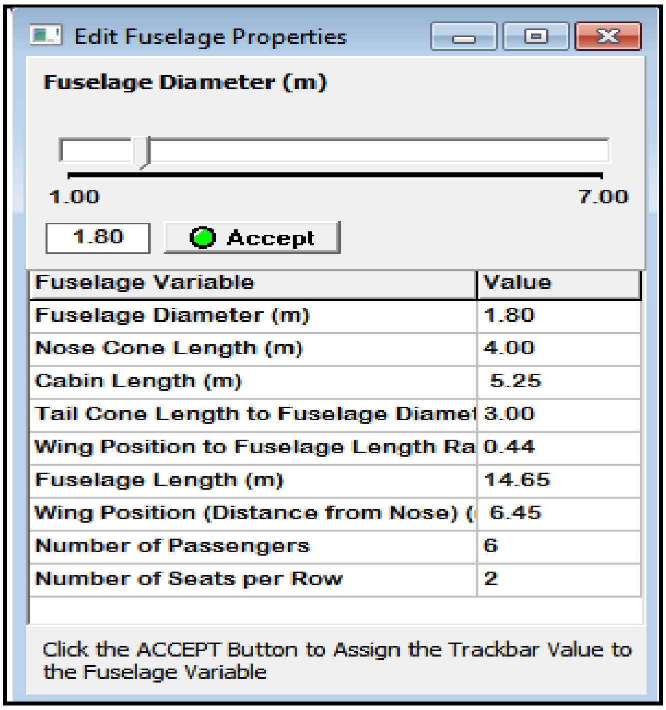
Wing Aspect Ratio

3.0 15.0

9.45 Accept

Wing Variable	Value
Wing Aspect Ratio	_9.45
Wing Taper Ratio	0.22
Wing Thickness Ratio	0.13
Wing SweepBack Angle (Deg.)	25.00
Wing Incidence Angle (Deg.)	3.00
Wing (Root to Average) Thickness F	1.10
Wing Span (m)	34.37
Wing Reference Area (m^2)	125.00
Wing Configuration	Low
Undercarriage Position	Wing

Figure A-2: Wing menu form



Edit Fuselage Properties

Fuselage Diameter (m)

1.00 7.00

1.80 Accept

Fuselage Variable	Value
Fuselage Diameter (m)	1.80
Nose Cone Length (m)	4.00
Cabin Length (m)	5.25
Tail Cone Length to Fuselage Diamet	3.00
Wing Position to Fuselage Length Ra	0.44
Fuselage Length (m)	14.65
Wing Position (Distance from Nose) (	6.45
Number of Passengers	6
Number of Seats per Row	2

Click the ACCEPT Button to Assign the Trackbar Value to the Fuselage Variable

Figure A-3: Fuselage menu form

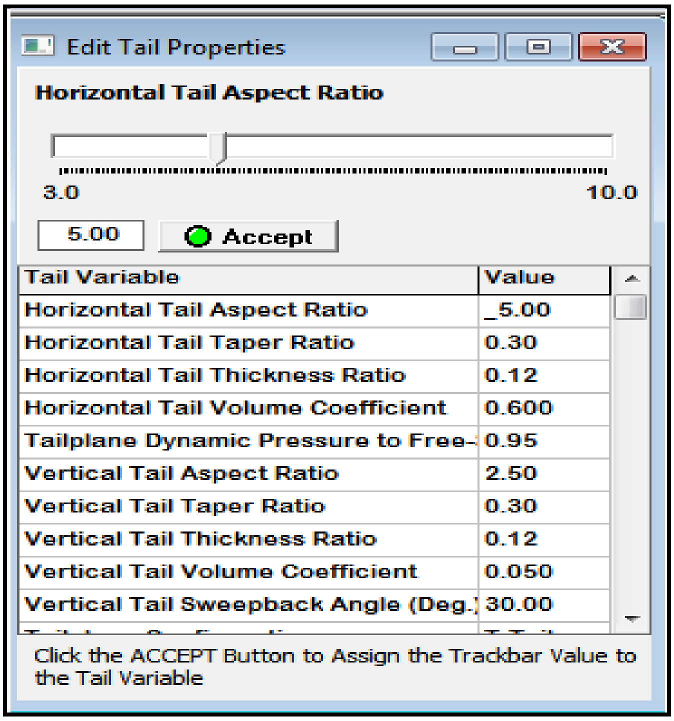


Figure A-4: Tail menu form

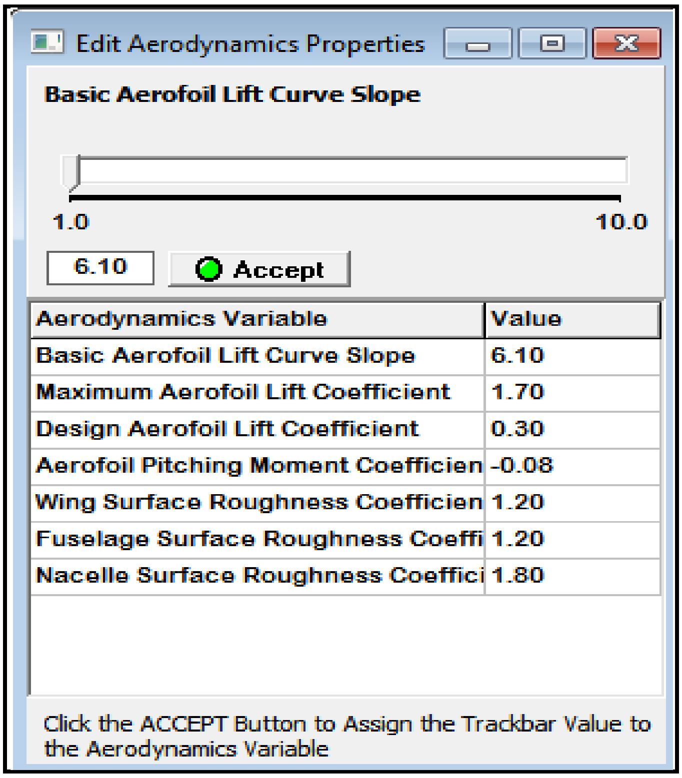


Figure A-5: Aerodynamic menu form

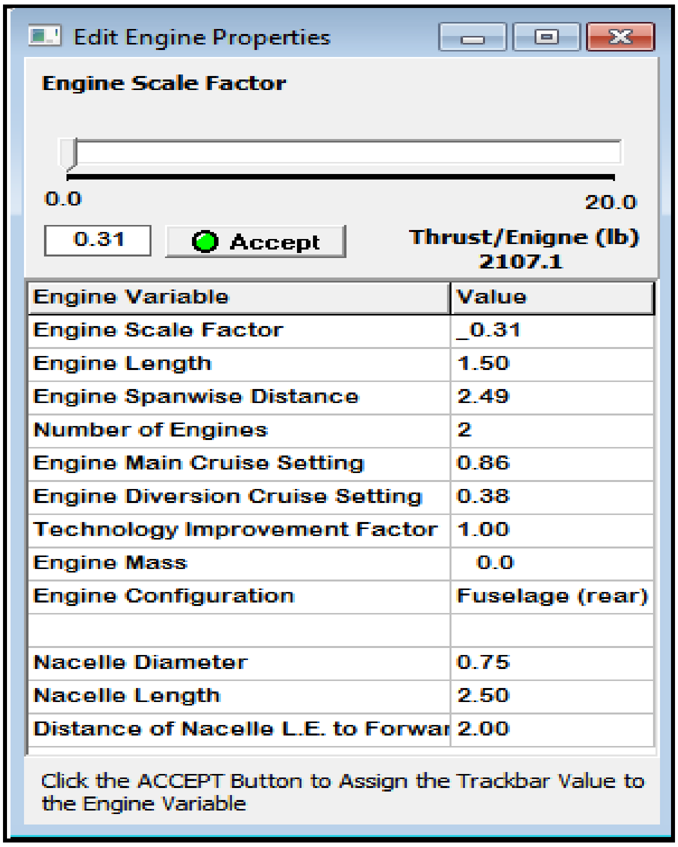


Figure A-6: Engine and Nacelle menu form

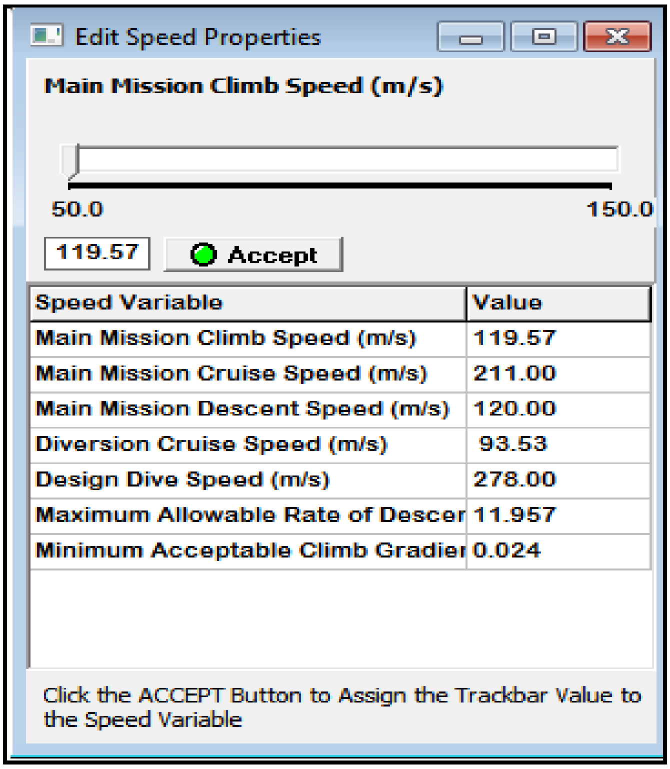


Figure A-7: Speeds menu form

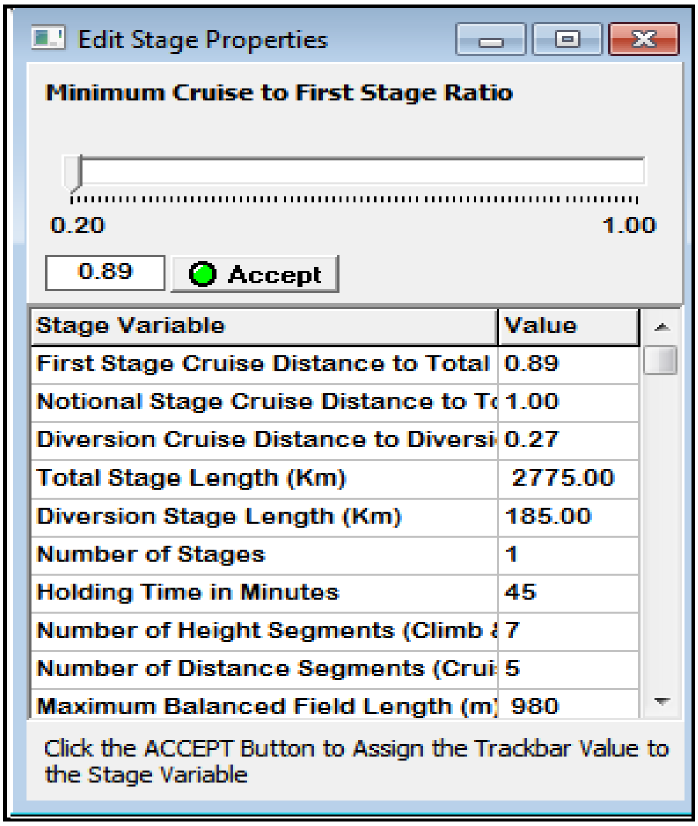


Figure A-8: Stages menu form

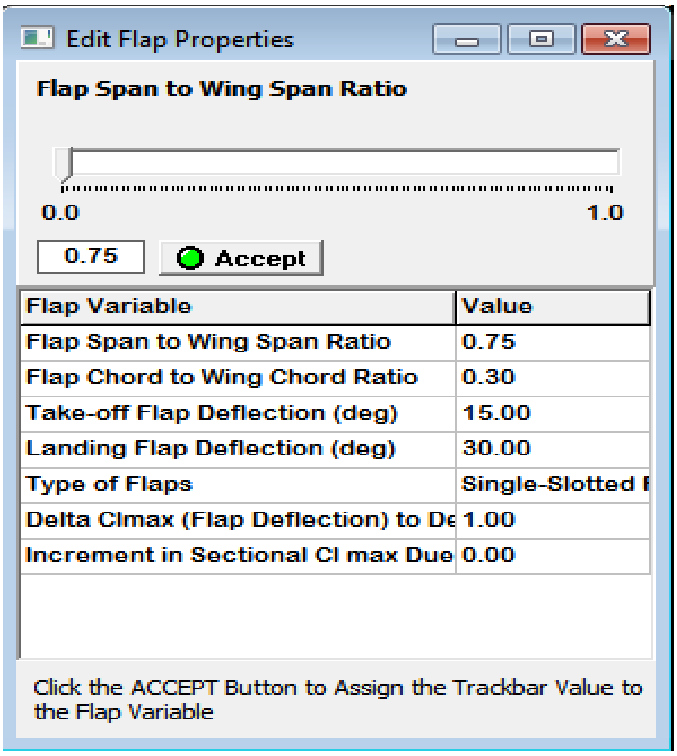


Figure A-9: Flaps menu form

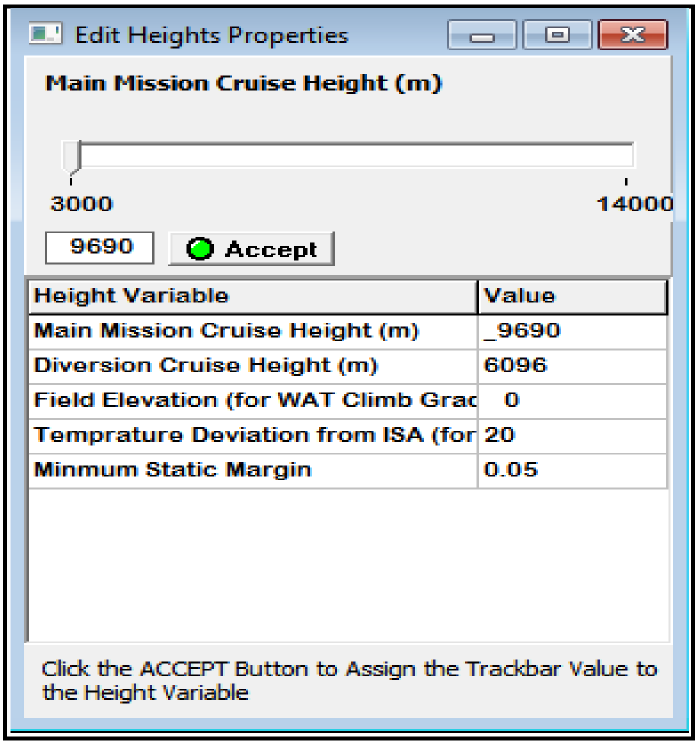


Figure A-10: Heights + WAT menu form

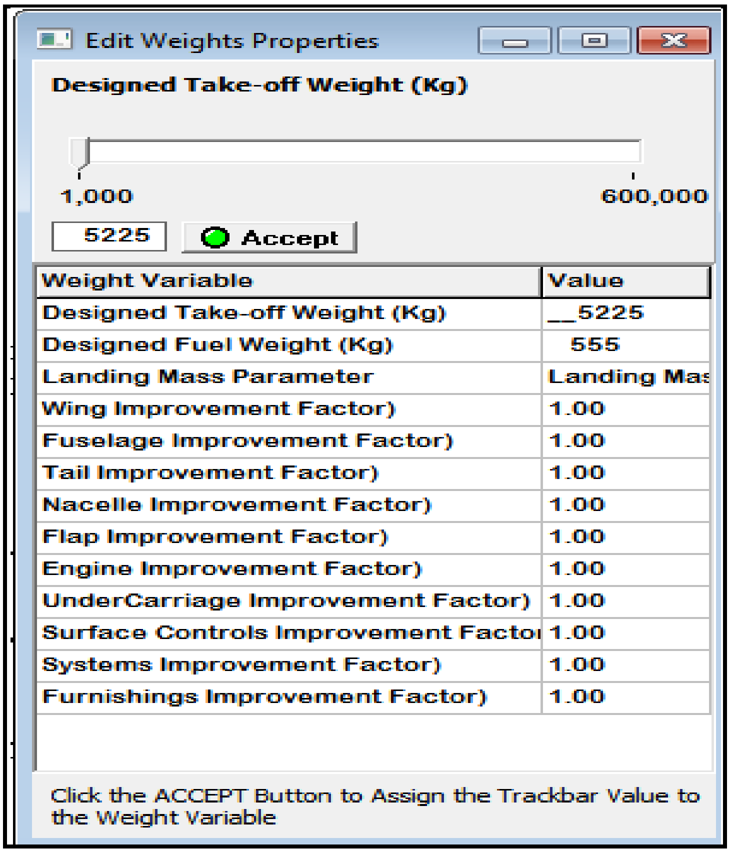


Figure A-11: Weights menu form

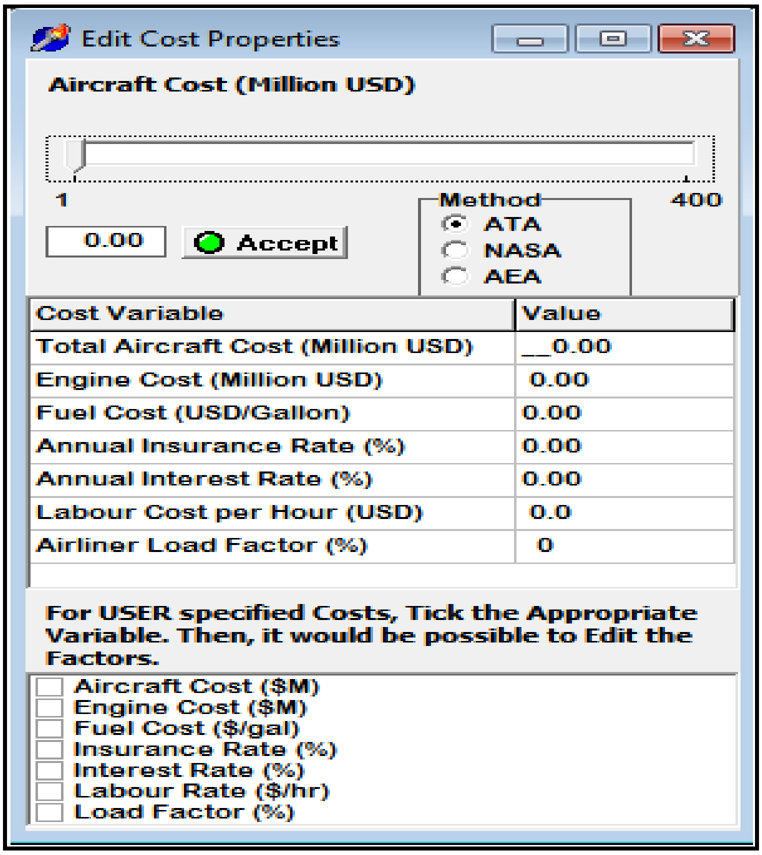


Figure A-12: Costs menu form

The next stage is to execute preliminary design using the *Design* menu in the main form as in **Figure A-13**. The first item *'Without optimisation'* performs the synthesis as a single pass. Selecting this item will open a form which contains all design variables and output parameters in tree structure as in **Figure A-14**. The benefit of using tree structure is to focus on certain parameters rather than a whole list. Hence, the user can change any design variable to see its effect on the selected output parameter easily and directly.

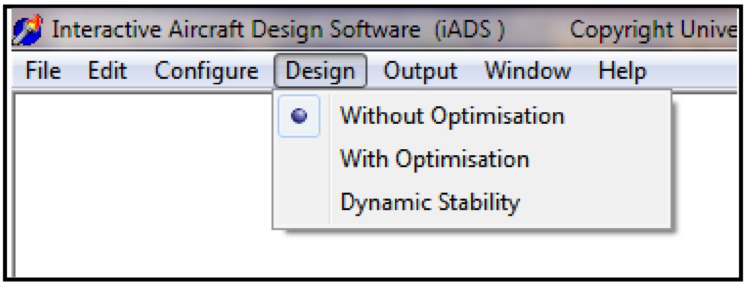


Figure A-13: Design menu form

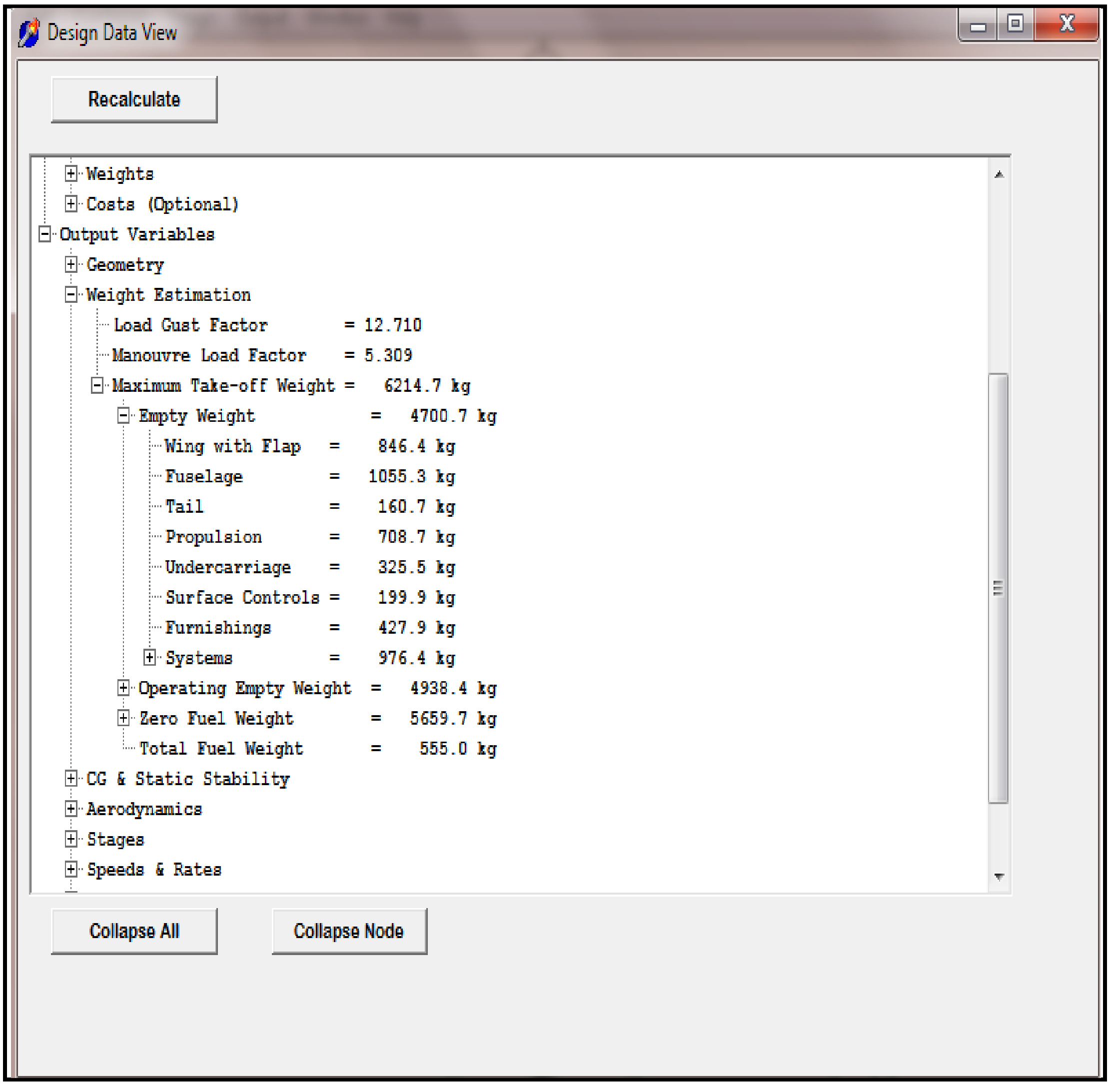


Figure A-14: Without optimisation output form

The second item in '*Design*' menu is '*With optimisation*', which activates the optimiser to perform optimisation for the designed aircraft. At the end of optimisation process, an alert message is displayed as in **Figure A-15**.

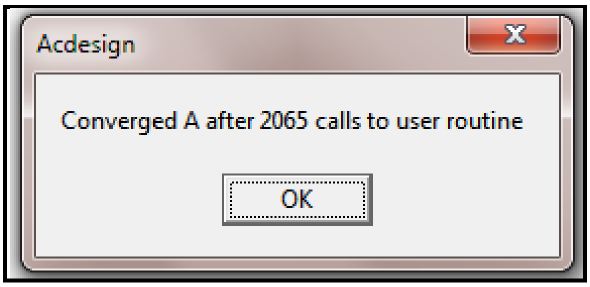


Figure A-15: Alert message form

It followed by another message to inform the end of optimisation process as in **Figure A-16**:

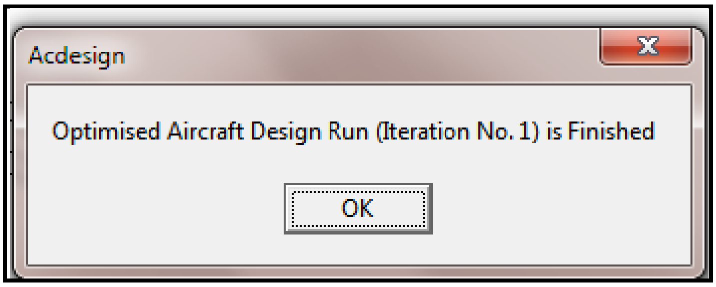


Figure A-16: End of optimisation process message form

The text output will be shown in '*Text Data*' item of '*Output*' menu.

The third item in *'Design'* menu is the *'Dynamic stability'*. It may be selected to perform the dynamic stability analysis as in **Figure A-17**. The user has a choice to select longitudinal or lateral analysis. Other buttons in the form are not activated until '*Calculate*' button is pressed. **Figure A-18** shows the form after longitudinal analysis is selected and '*Calculate*' button has been pressed. To plot the response, the user can select the response mode and the period of plot and press '*Plot*' button as in **Figure A-19**.

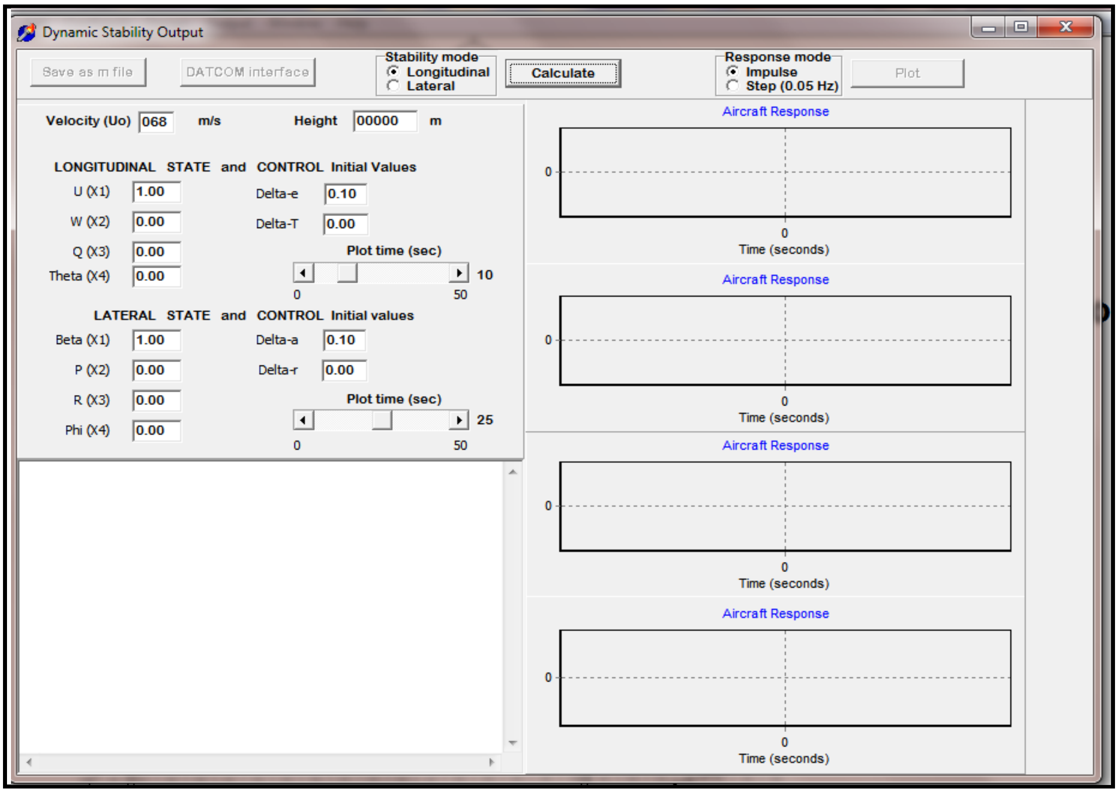


Figure A-17: Dynamic stability form

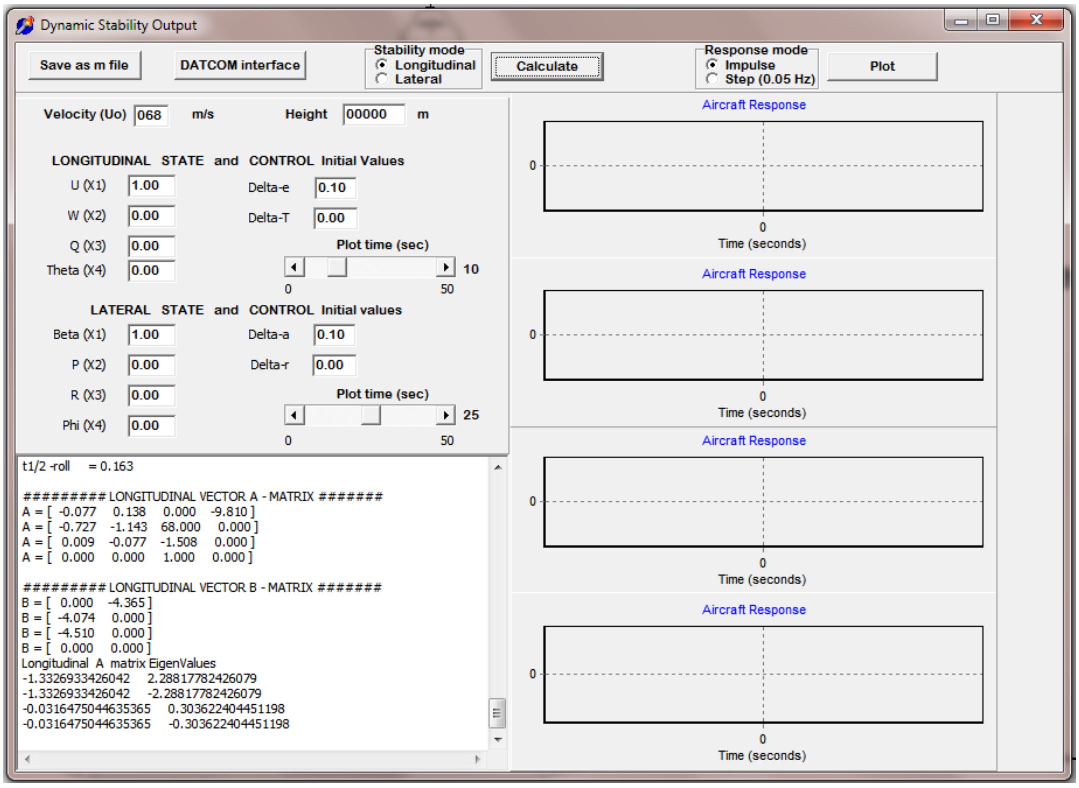


Figure A-18: Longitudinal analysis output

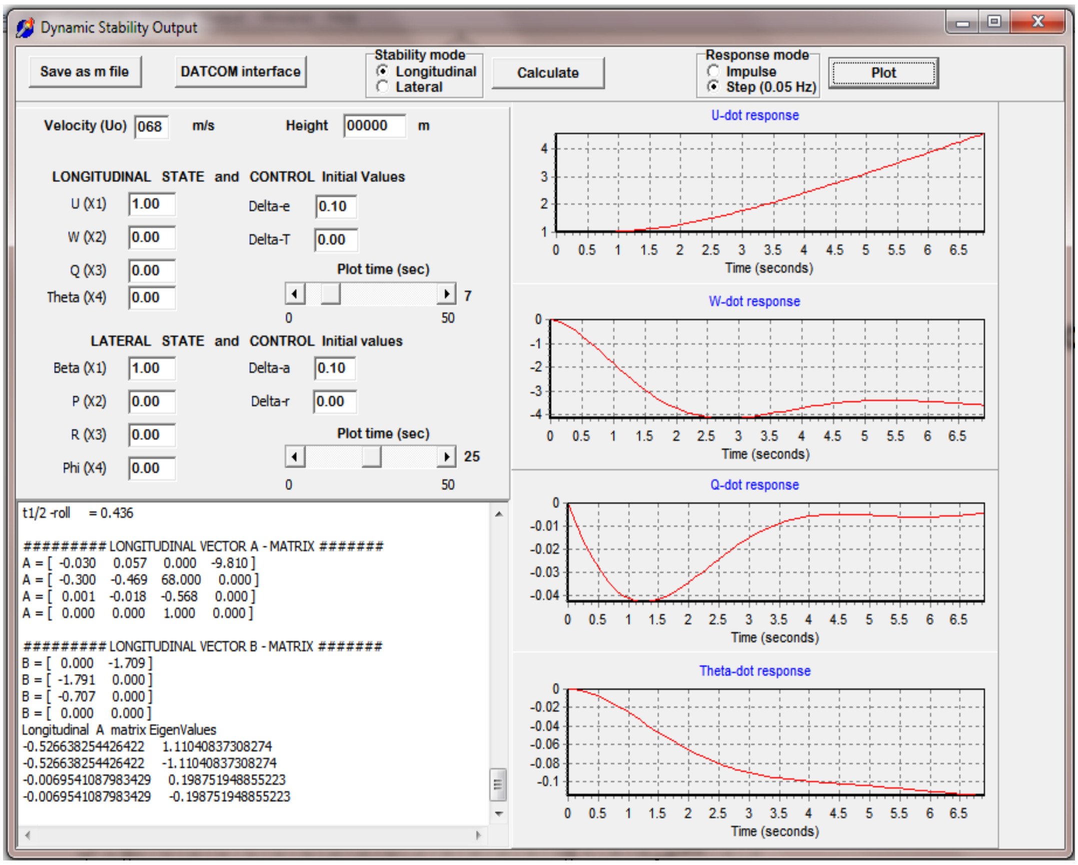


Figure A-19: Longitudinal analysis and plot form

The '*DATCOM* interface' button is used to create the input file for DATCOM software. **Figure A-20** shows the DATCOM interface form. It contains a tabbed notebook of six elements. The first element '*Flight condition*' is used to define the number of Mach number, number of AOA, number of heights, and their values used for analysis as shown in **Figure A-21**. *'Save'* button will be activated after all data are entered.

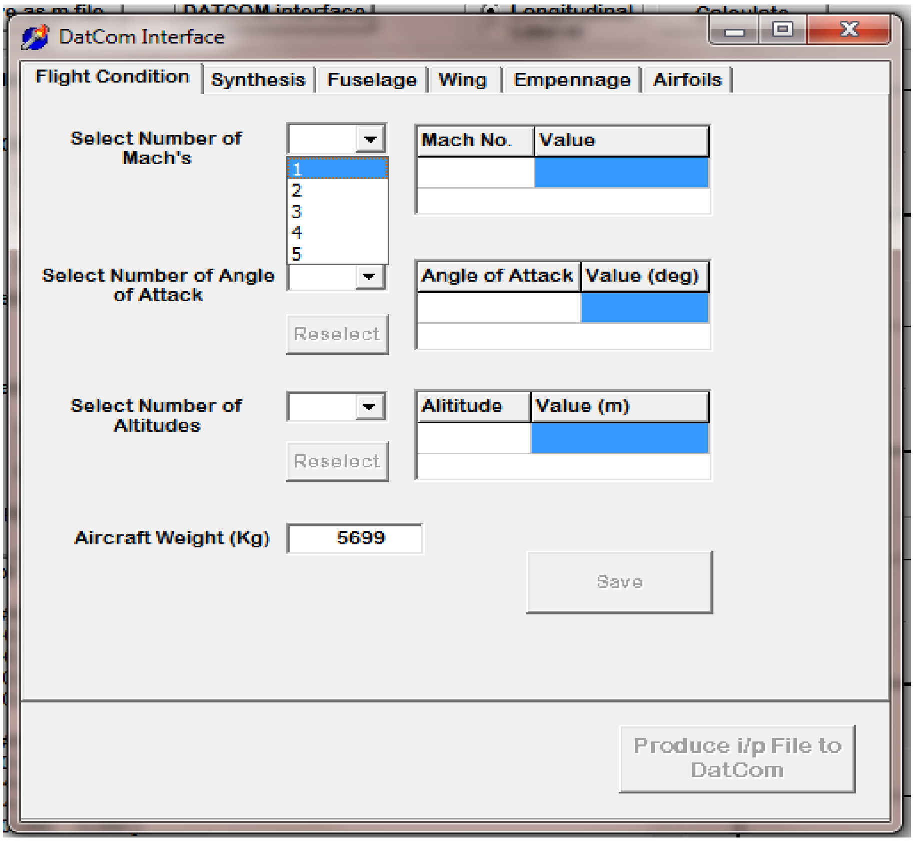


Figure A-20: DATCOM interface form

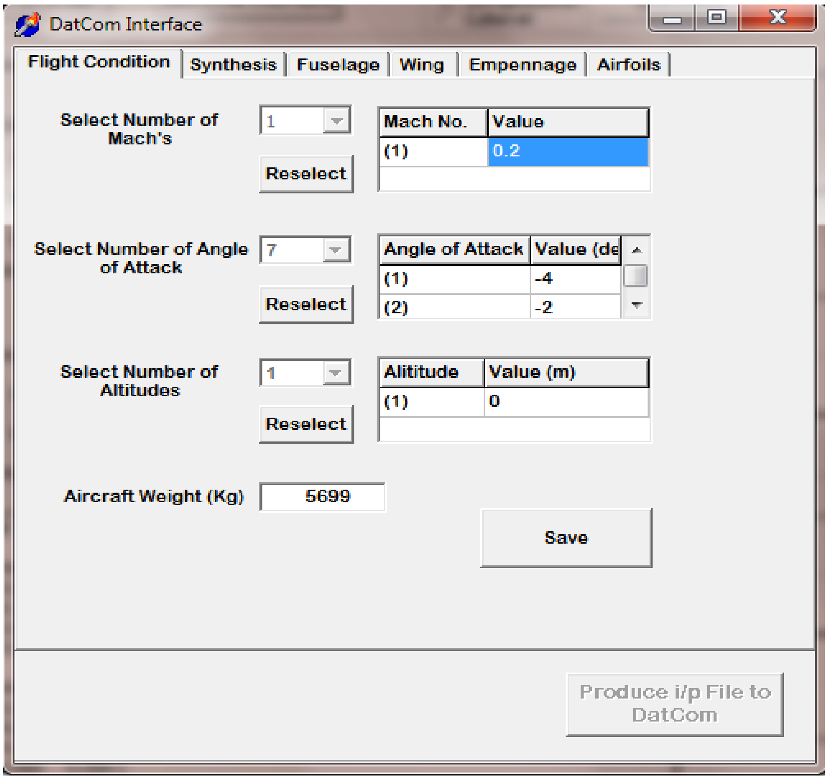


Figure A-21: Flight condition form

The second element is '*Synthesis*' contains ten variables where their values are passed from the main program as in **Figure A-22**. The user has the opportunity to accept these values or enter his value. '*Save'* button commits these values for incorporation in the DATCOM interface file.

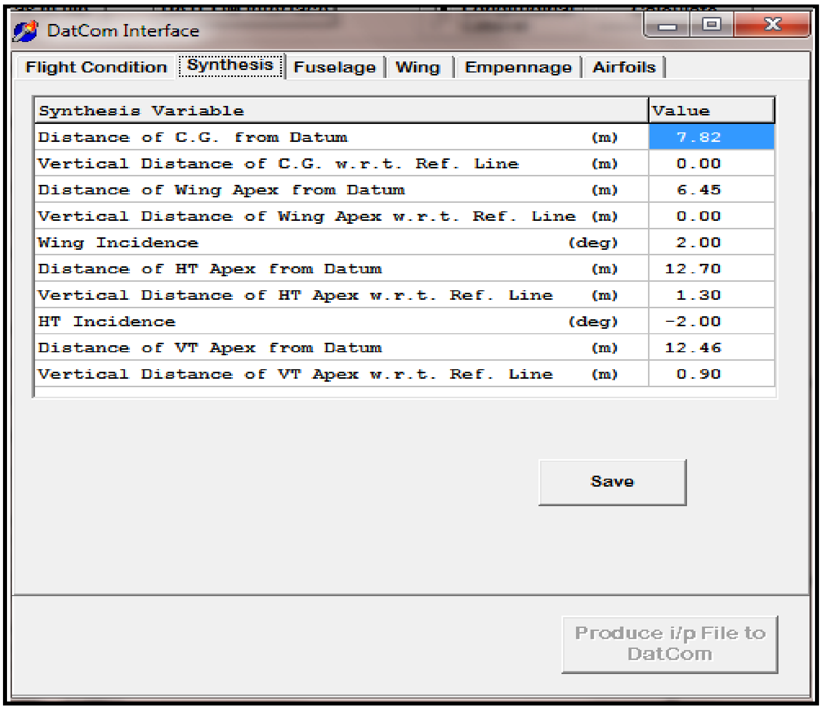


Figure A-22: Synthesis form

'*Fuselage*' is the third element in DATCOM interface. The user should select the number of fuselage stations under consideration. Each station has two values, one for distance and the other is for cross sectional area. Again, all information input must be saved, the '*Save'* button must be pressed after entering all values. **Figure A-23** shows 'Fuselage' element form:

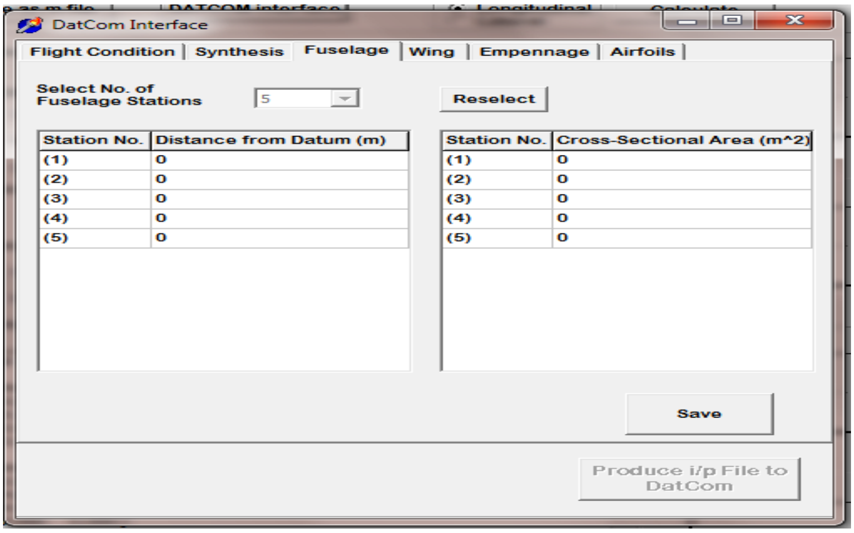


Figure A-23: 'Fuselage' form

All variables required for '*Wing*' element are passed from the main program as shown in **Figure A-24**. '*Save'* button commits these values for incorporation in the DATCOM interface file.

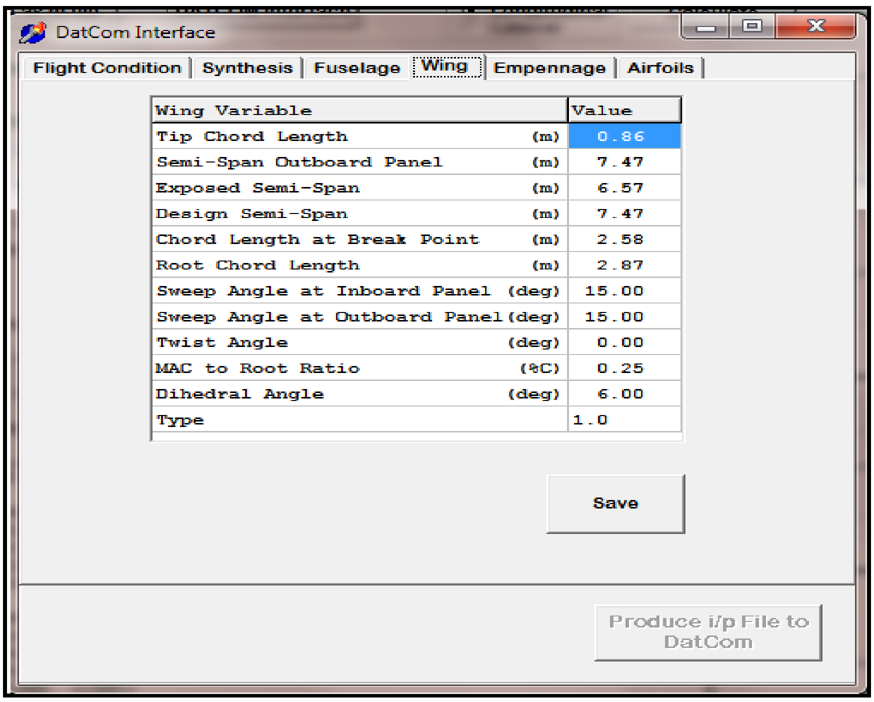


Figure A-24: 'Wing' form

Empennage form is similar to wing form which contains variables that their values are passed from the main program as in **Figure A-25**.

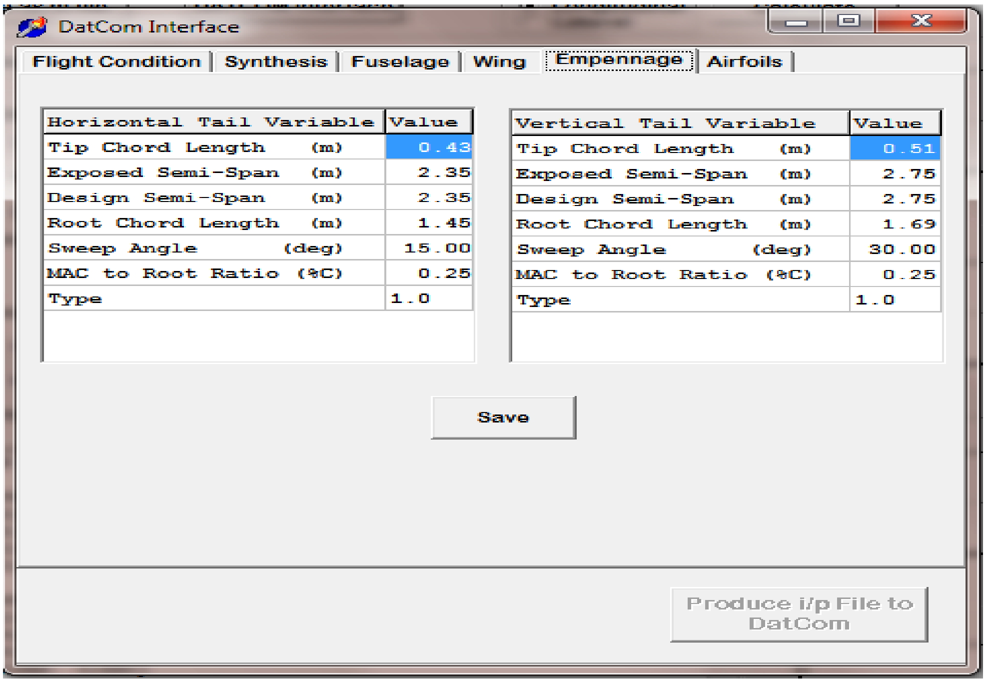


Figure A-25: 'Empennage' form

The last element is the '*Airfoil*' which is used to enter wing and tail aerofoil numbers. Aerofoil series are selected first and its number is entered in the text box. Save button will activate after all data are entered as in **Figure A-26**. The '*Produce i/p File DatCom*' button is activated after all *'Save'* buttons for the six elements have been pressed.

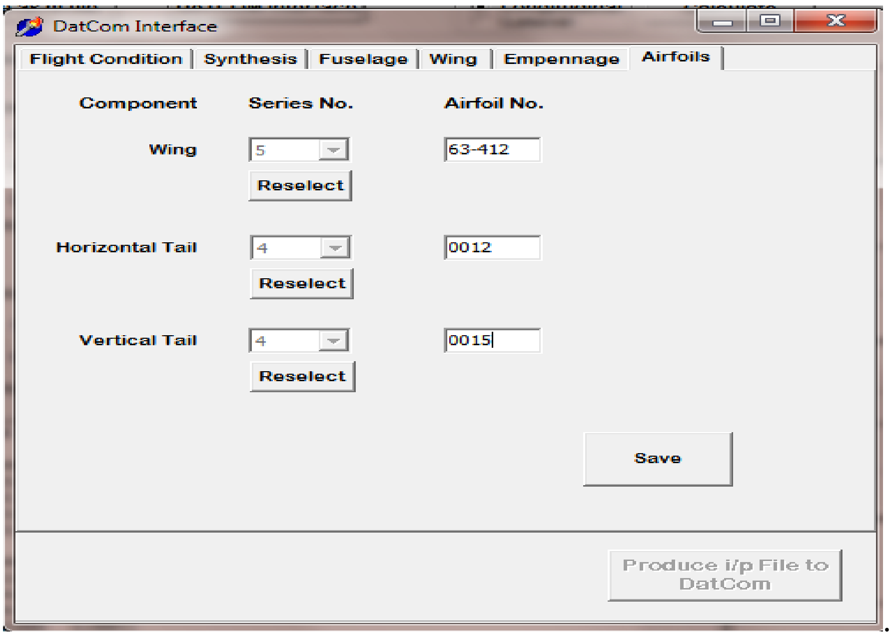


Figure A-26: Aerofoil form

The '*Output*' item in the main menu form contains three elements as in **Figure A-27**.

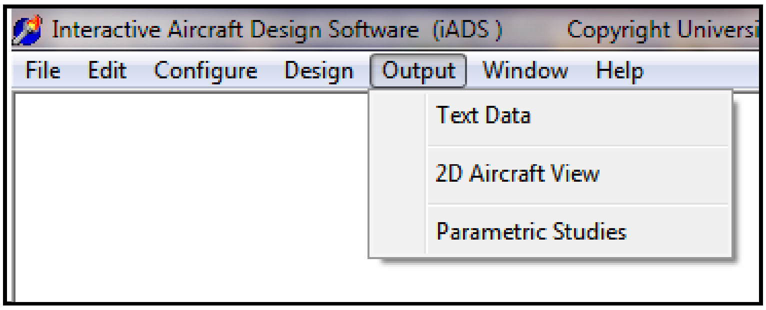


Figure A-27: Output menu form

'Text Data' element shows the output text list: for the synthesis program as in Appendix II, for the optimiser as in Appendix VI, and for the takeoff module as in **Figures A-28, A-29, & A-30**, respectively. Clicking on the labels at the top will activate the required form. The ability to save data in an Excel spreadsheet, as well as a text file, is another feature for iADS software.

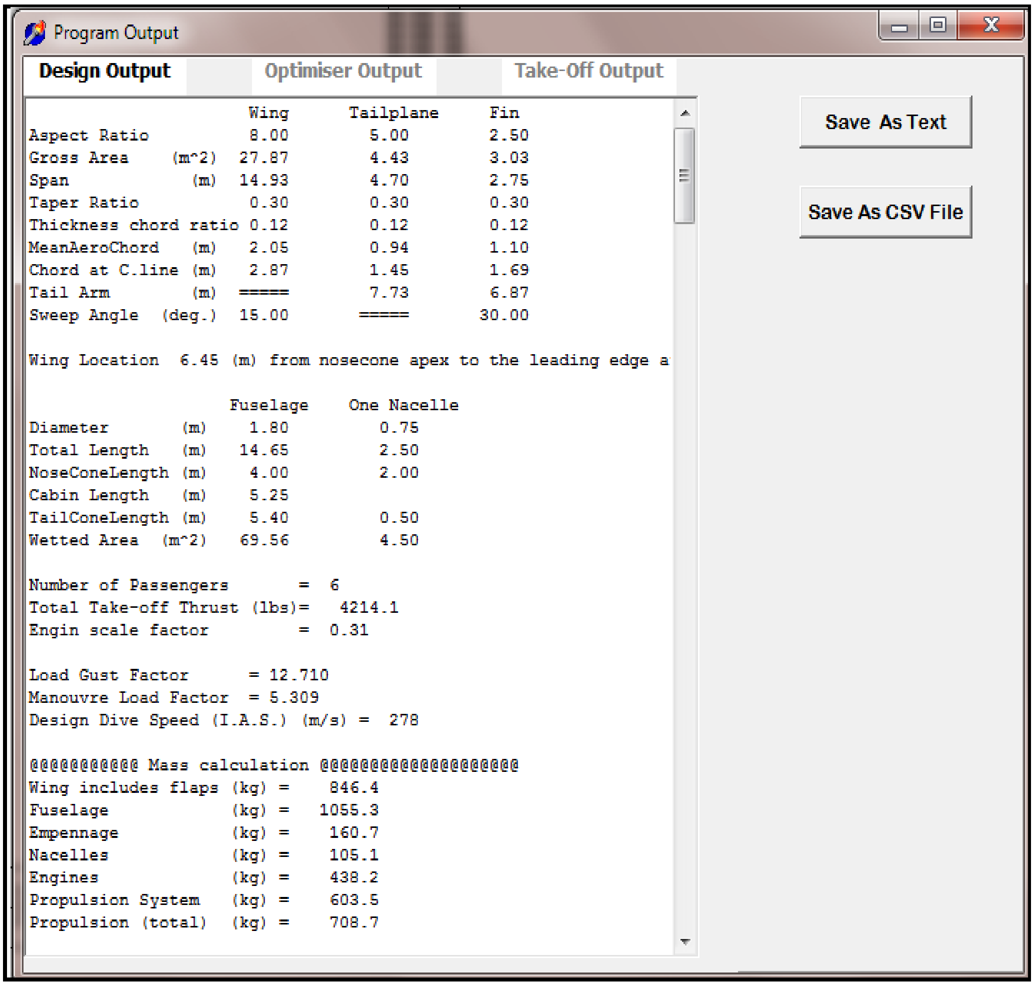


Figure A-28: Text output form for the synthesis program

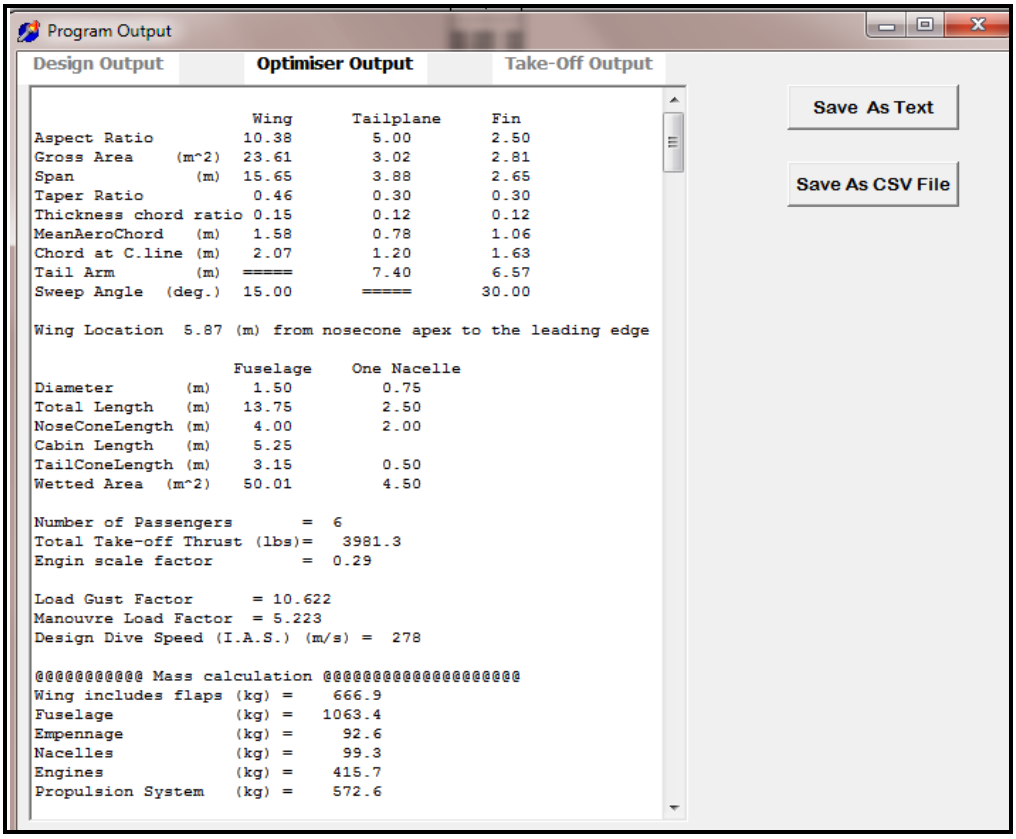


Figure A-29: Text output form for the optimiser

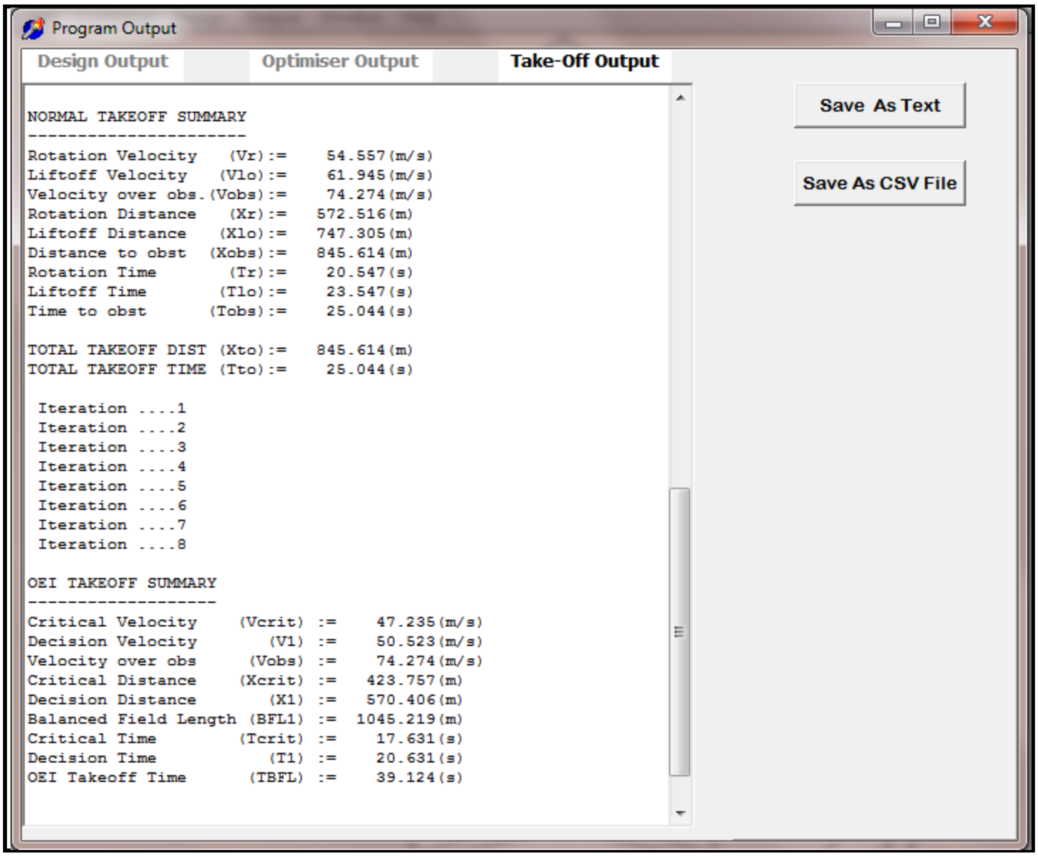


Figure A-30: Text output form for the takeoff module

‘*2D Aircraft View’* element shows the 3-view of the designed aircraft as in **Figure A-31**.

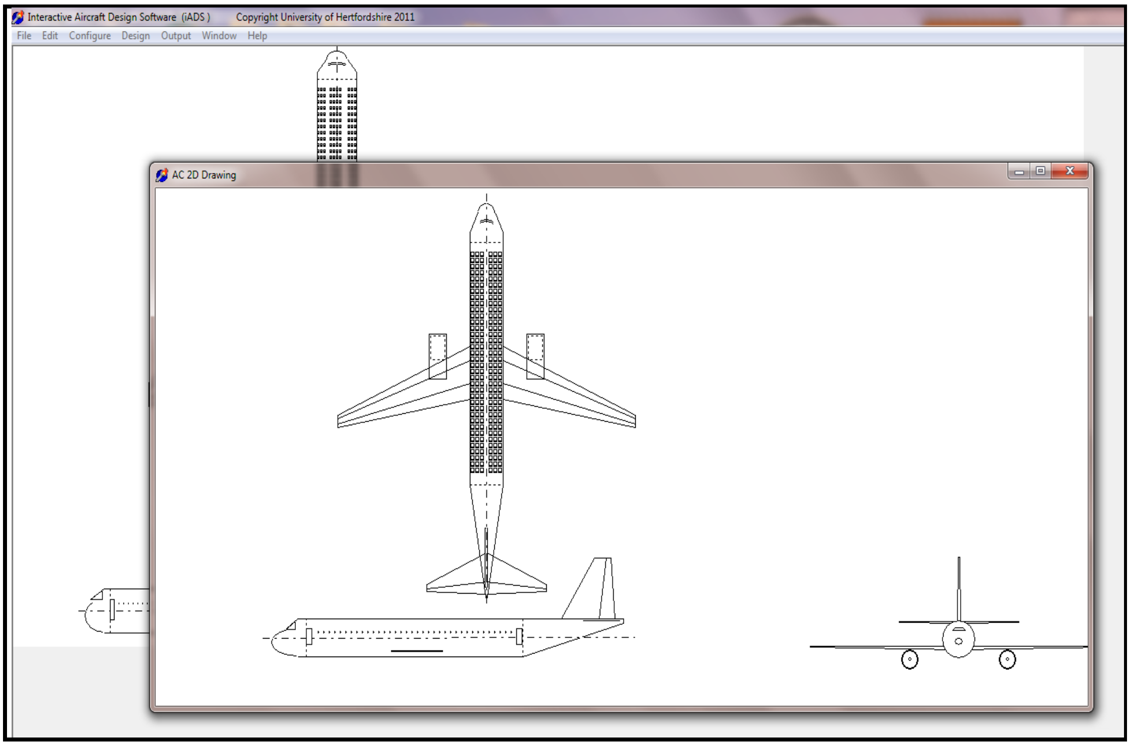


Figure A-31: 3-View output form

The user has the opportunity to resize the form to his need without losing any information, as in **Figure A-32**:

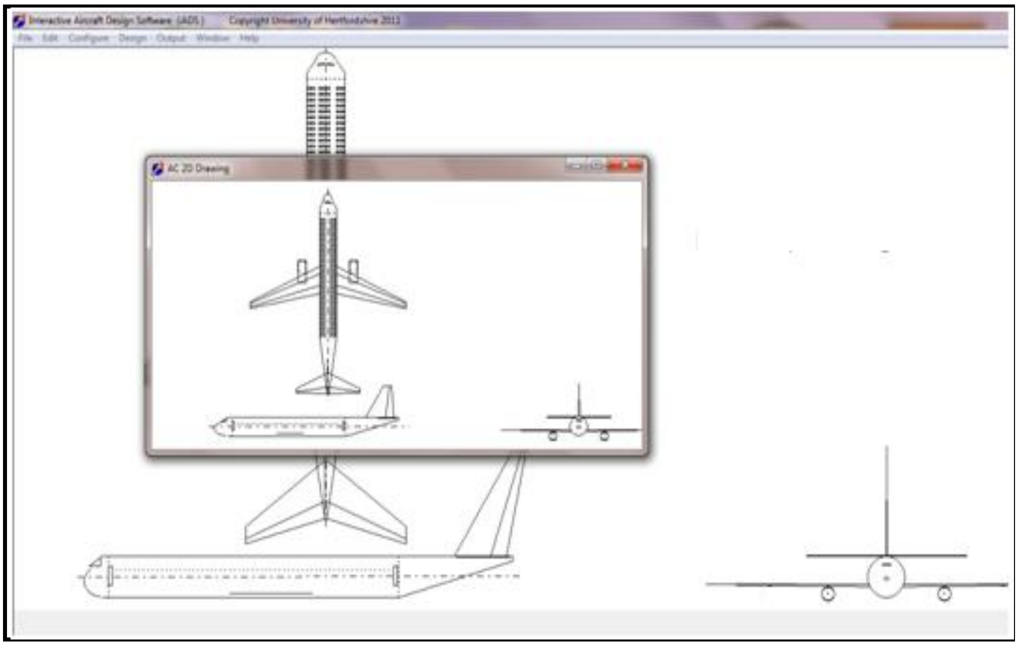


Figure A-32: Resizable draw form

This feature is very valuable in configuring the designed aircraft. In entering any variable related to the aircraft geometry, this form is launched automatically as an OOUI application as shown in **Figure A-33**. This allows students to understand the effect of the design variable graphically.

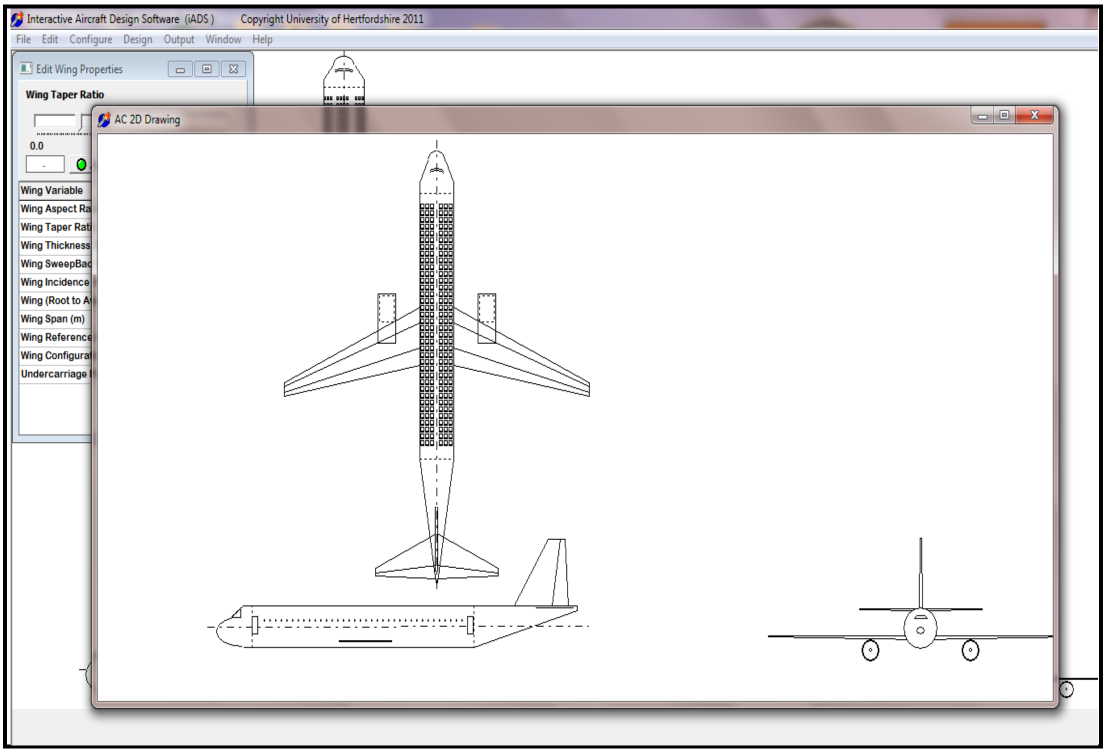


Figure A-33: OOUI application

Figure A-34 shows the application after resizing:

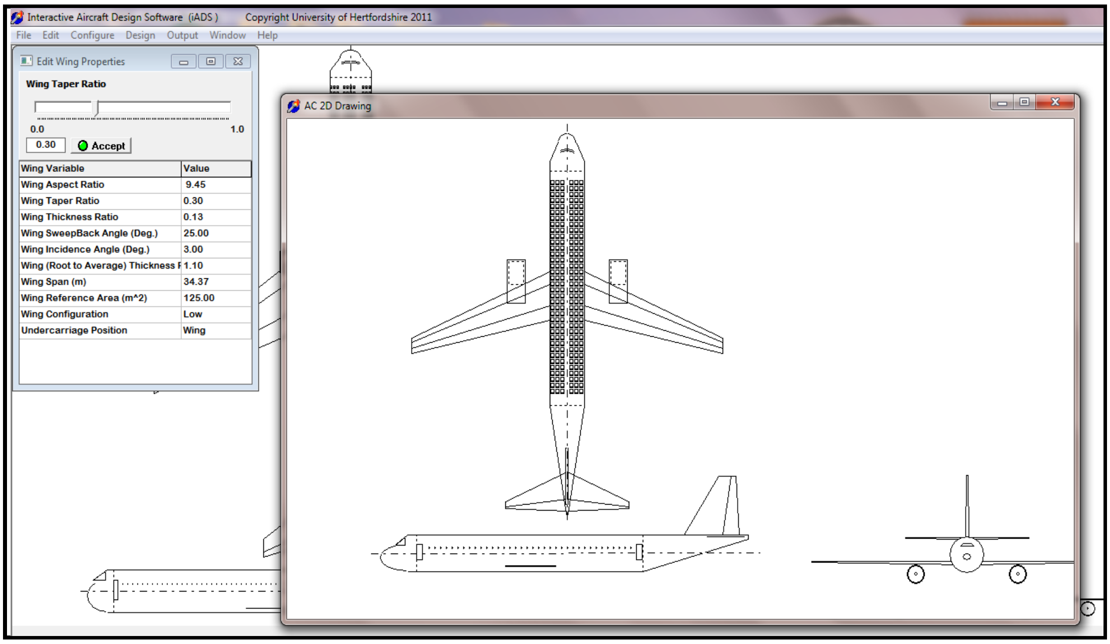


Figure A-34: Resizable OOUI application

The last element in '*Output*' menu is 'Parametric Studies'. This module has great benefit to students enabling them to investigate the effect of design variables variation on the output parameters. **Figure A-35** shows the parametric studies form.

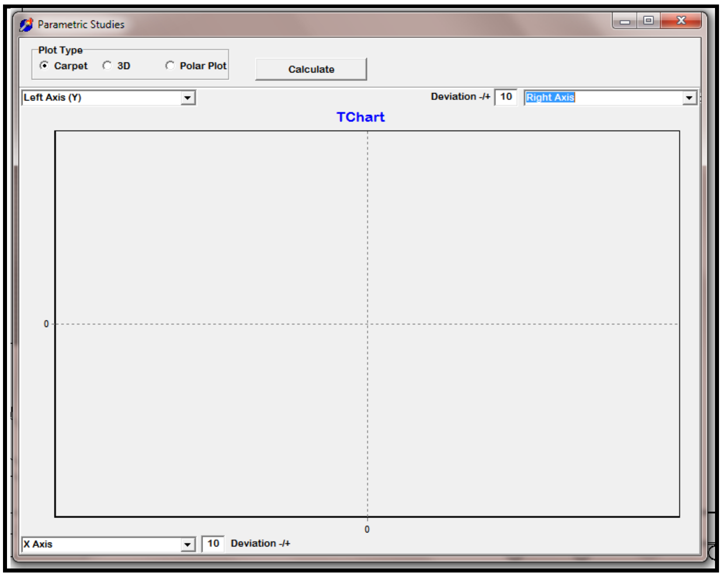


Figure A-36: Parametric studies form

Initially, the user can select the output parameter under consideration from the drop list of the Y axis as in **Figure A-37**.

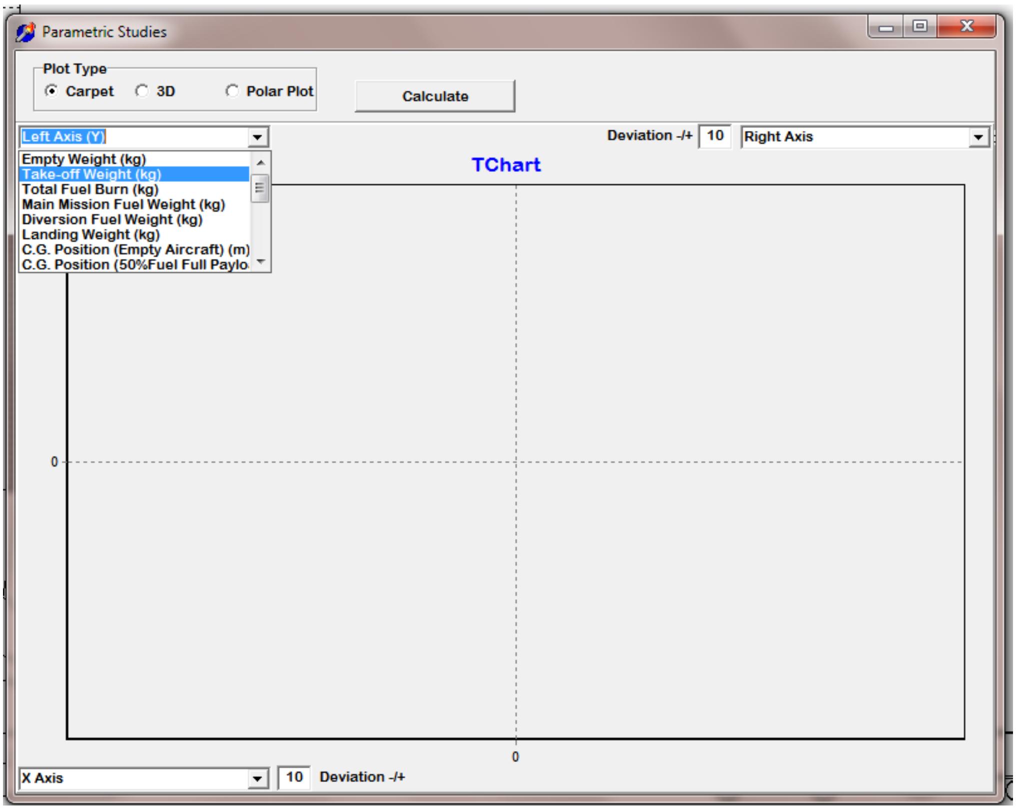


Figure A-37: Drop list in parametric studies form

X axis is selecting similar to Y axis. Press '*Calculate*' to show the plot as in **Figure A-38**. The deviation field, next to each axis, is added to allow the user to select the range of the variable to be changed.

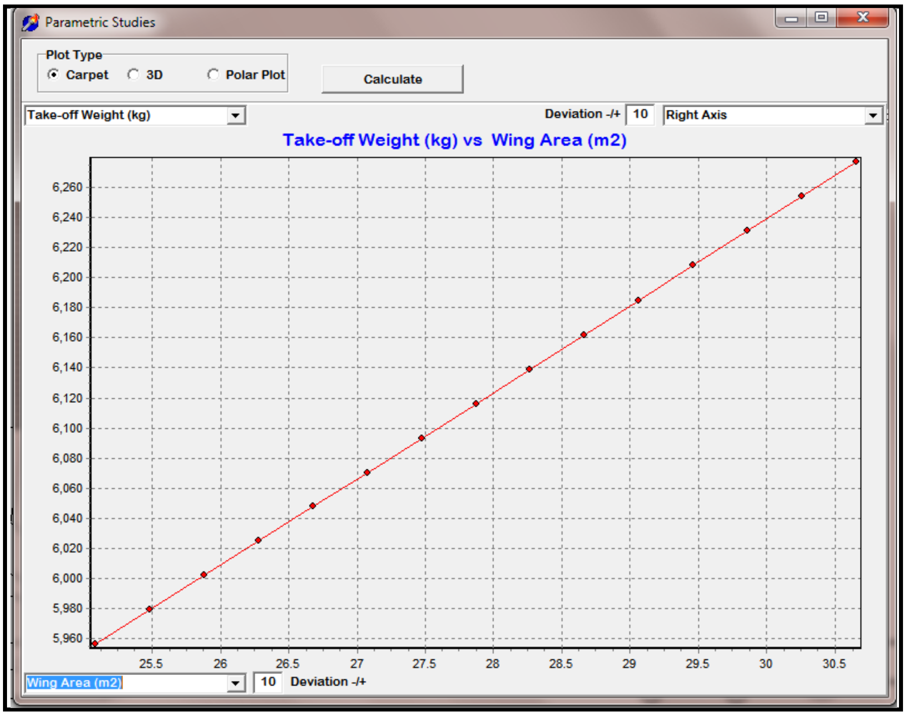


Figure A-38: 'One-to One' 2D plot form

Selecting the right axis, a plot of 'One –to-Two' is shown as in **Figure A-39**. Note that the line with dot points represents the lowest value for the right axis.

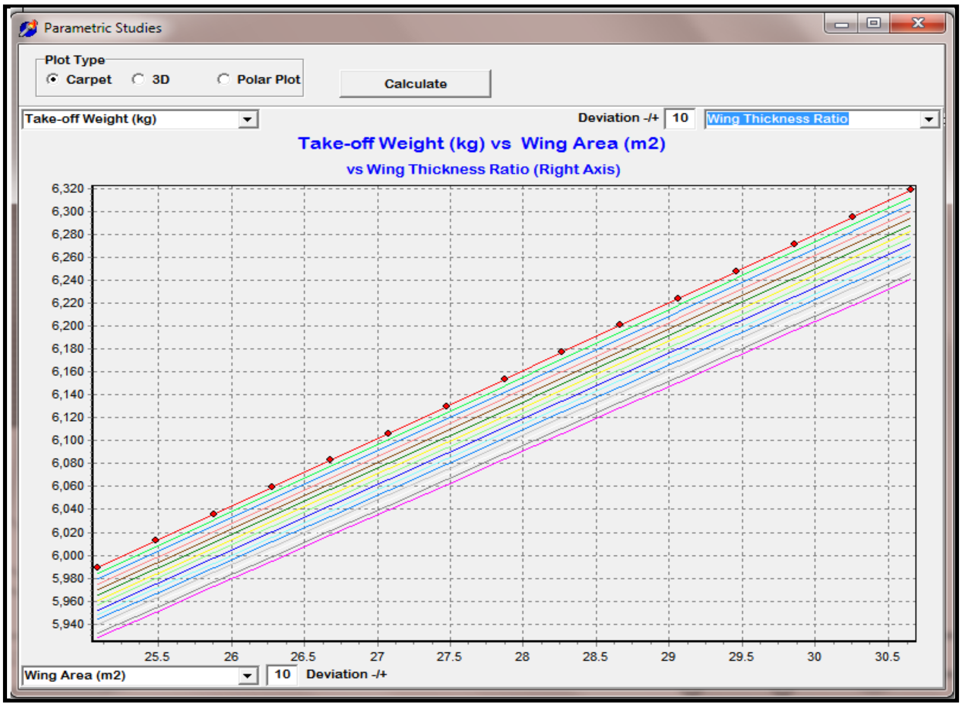


Figure A-39: 'One-to-Two' 2D plot form

The opportunity to select 3D plot is also available in this form as in **Figure A-40**.

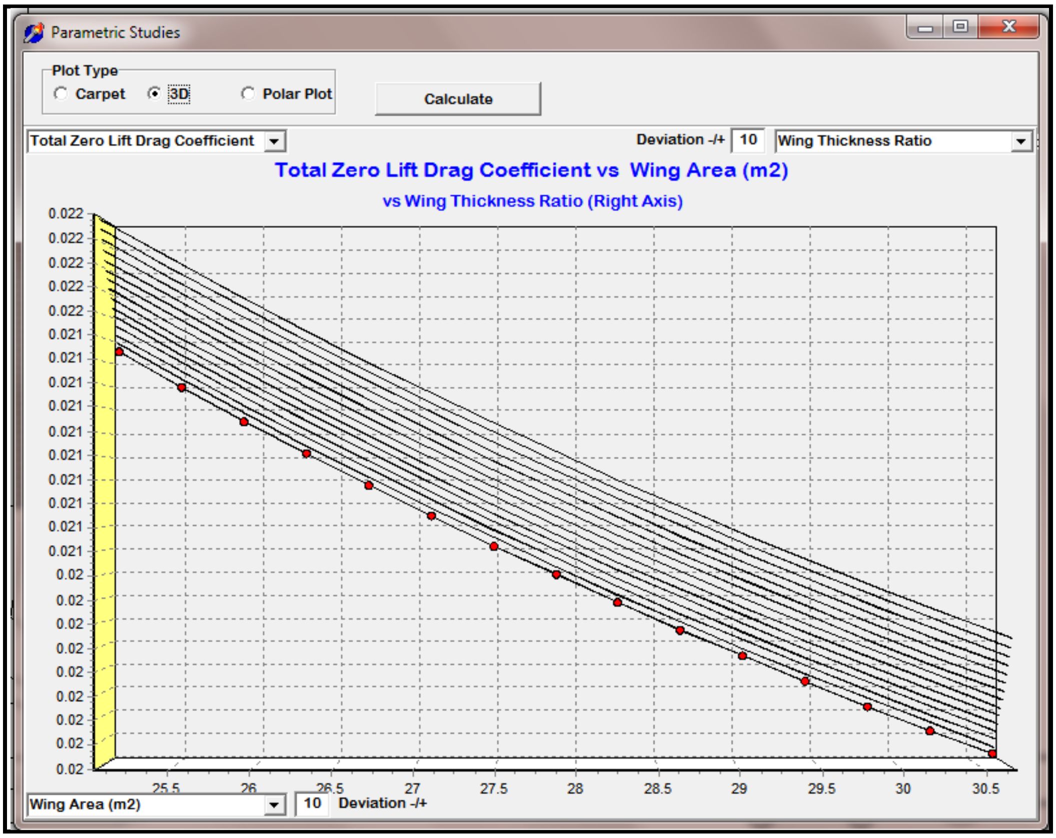


Figure A-40: ‘One-to-Two’ 3D plot form

The third choice in parametric studies form is the polar plot as in **Figure A-41**.

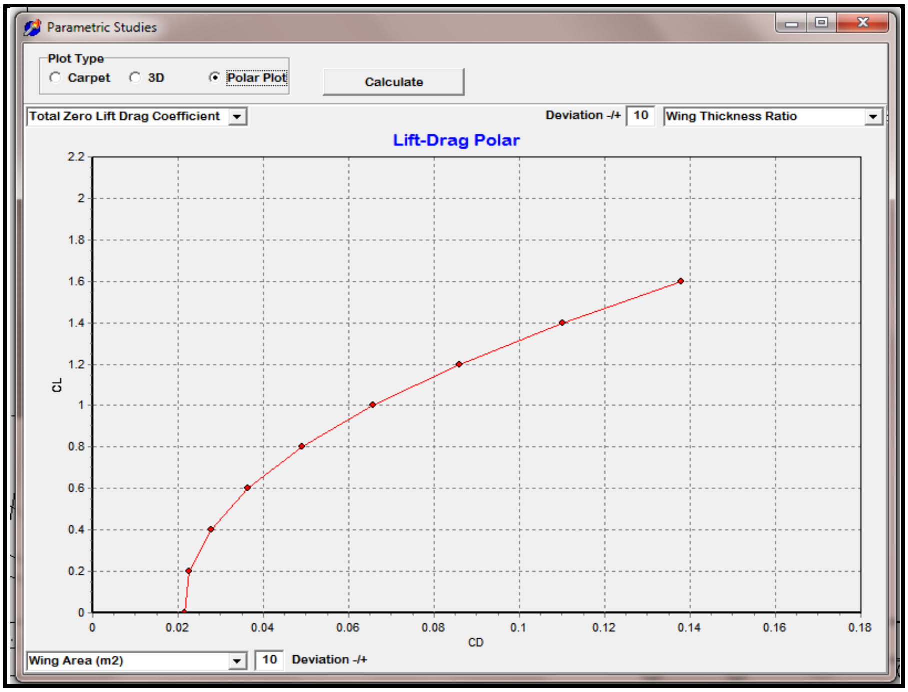


Figure A-41: Polar plot form

Parametric studies form is also resizable as shown in **Figure A-42**.

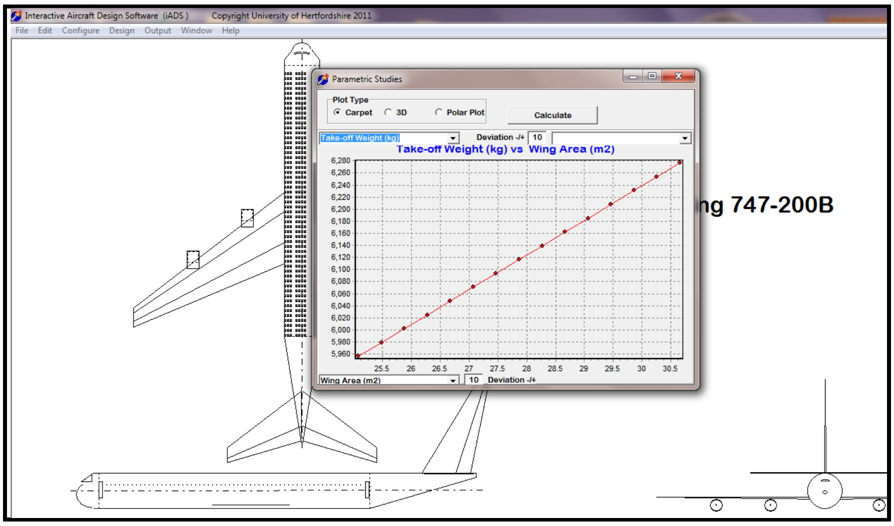


Figure A-42: Resized Parametric studies form

Additional feature was added to iADS software which is '*Data Base*' folder. This folder contains all the necessary design data of the existing Airbus and Boeing aircraft. To select an Airbus aircraft, from '*File*' menu in the main form of iADS, select '*open*' to show the input window. In the '*file name*' field just press 'a', a drop list is opened to select the desired Airbus aircraft as in **Figure A-43**.

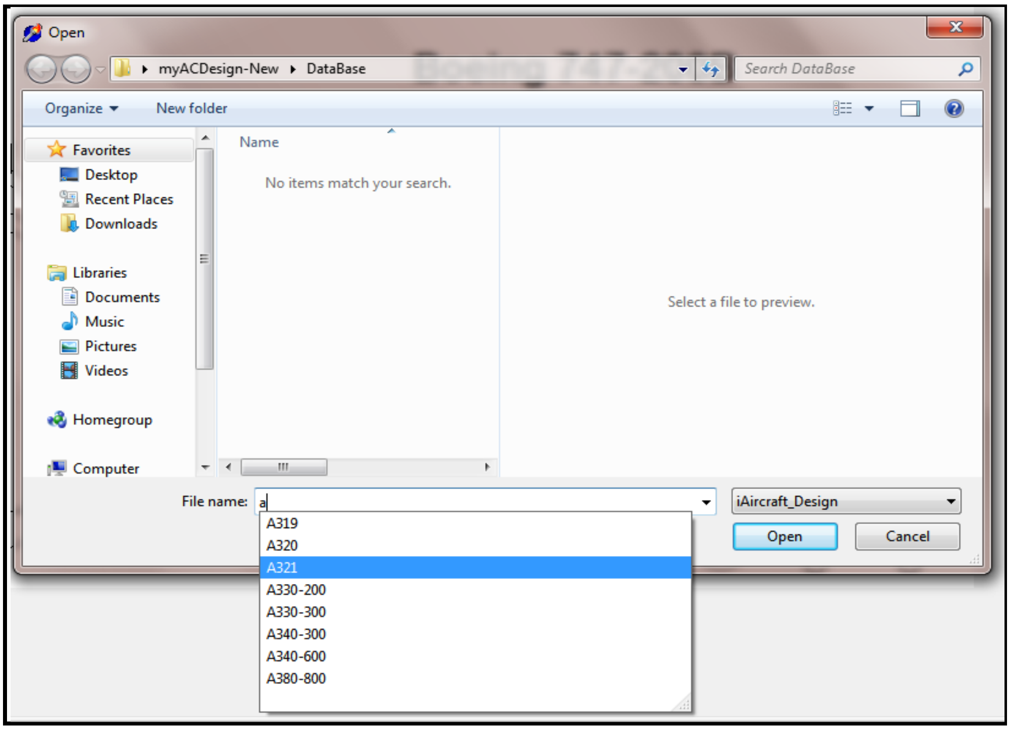


Figure A-43: Airbus aircraft selection form

In similar manner, the user can select a Boeing aircraft by entering number '7' instead of 'a' in the file name as in **Figure A-44**.

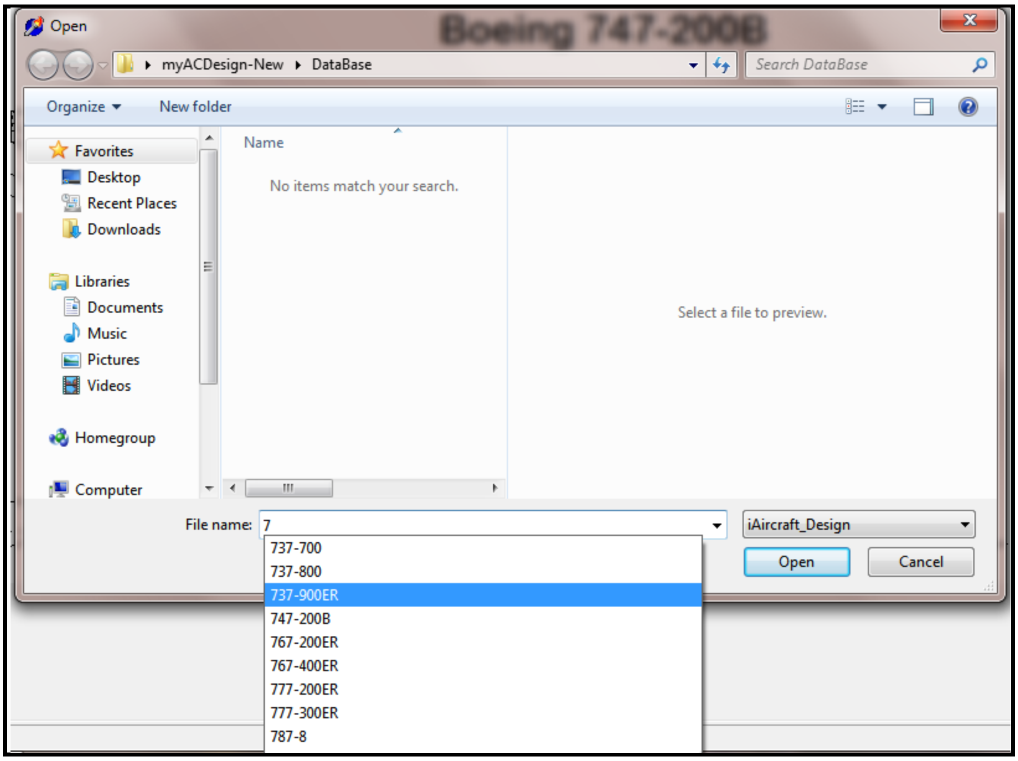


Figure A-44: Boeing aircraft selection form

Finally, a '*Takeoff*' module is added to iADS to analyse the takeoff stage in detail. '*Takeoff setting'* item in '*Configure*' menu is the input form for its design variables needed as in **Figure A-45**. Although, most of these values are passed directly from the main program, the user has the opportunity to change these data. The text output of this module is shown in **Figure A-30** as mentioned before.

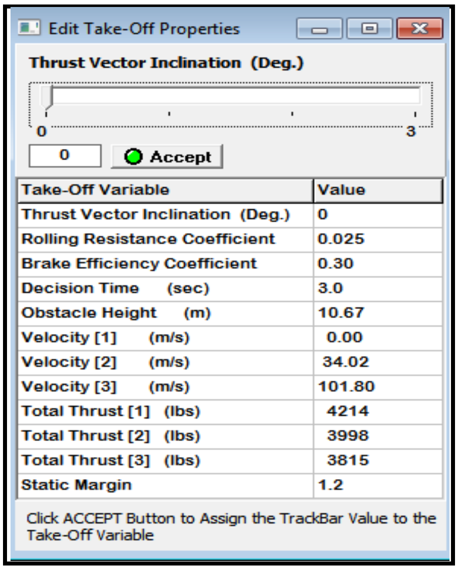


Figure A-45: Takeoff setting form