SEPTEMBER 2018 M. Sc. Thesis in Aircraft and Aerospace Engineering SAEED BAROUD

UNIVERSITY OF GAZİANTEP

GRADUATE SCHOOL OF

NATURAL & APPLIED SCIENCE

DESIGN OF A USER-FRIENDLY SOFTWARE FOR AIRCRAFT PERFORMANCE ESTIMATION

M. Sc. THESIS

IN

DEPARTMENT OF AIRCRAFT AND AEROSPACE ENGINEERING

BY

SAEED BAROUD

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Design of a User-Friendly Software

for Aircraft Performance Estimation

M.Sc. Thesis

in

Aircraft and Aerospace Engineering

University of Gaziantep

Supervisor

Assist. Prof. Dr. M. Veysel ÇAKIR

by

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This is to certify that we have read this thesis and that in our consensus/majority opinion it is fully adequate, in scope and quality, as a thesis for the degree of Master of Science.

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Saeed BAROUD

ABSTRACT

DESIGN OF A USER-FRIENDLY SOFTWARE FOR AIRCRAFT PERFORMANCE ESTIMATION

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This thesis is concerned with the analysis of aircraft performance as a part of an aircraft design. It is an essential process to analyse the aircraft performance to compare whether it is matching with the performance expected from the aircraft. In this study, Aircraft performance estimation software was developed using the easy-to-use MATLAB GUI interface. The goal of this software is to have user friendly and open source features which are two main features. Because these two features enable the software to include more students, researchers, innovators and engineers in the aircraft design process, which enhances the innovation of newly developed aircraft. The software development process covers the mathematical modelling, program development, and verification of results. The accuracy of the software is determined by comparison with known aircraft performance characteristics. The thesis findings are summarized in the discussion and conclusion section.

Key Words: Aircraft design, aircraft performance, open-source software, UPA-Gaziantep

ÖZET

UÇAK PERFORMANS TAHMİNİ İÇİN KULLANICI-DOSTU BİR YAZILIMIN TASARLANMASI

BAROUD, Saeed
Yüksek Lisans Tezi, Havacılık Ve Uzay Mühendisliği
Tez Yöneticisi: Dr. Öğr. Üyesi M. Veysel ÇAKIR
Eylül 2018
91 sayfa

Bu tez, uçak tasarımının bir bölümü olan uçak performans analizi ile ilgilidir. Uçak performans analizi yapmak, uçaktan beklenen performansın gerçekleşip gerçekleşmeyeceğini kontrol etmek için gerekli bir süreçtir. Bu çalışmada kullanımı kolay MATLAB GUI ara yüzü kullanılarak uçak performans tahmin yazılımı geliştirilmiştir. Bu yazılımın amacı, iki ana özellik olan kullanıcı dostu ve açık kaynak özelliklerine sahip olmaktır. Çünkü bu iki özellik, yeni geliştirilen uçakların inovasyonunu arttıran, uçak tasarım sürecine, daha fazla öğrenci, araştırmacı, yenilikçi ve mühendisi dahil etmeyi mümkün kılacaktır. Yazılım geliştirme süreci, matematiksel modelleme, program geliştirme ve sonuçların doğrulanmasını kapsamaktadır. Yazılımın doğruluğu, bilinen uçak performans karakterleriyle karşılaştırılarak belirlenmektedir. Tez bulguları tartışma ve sonuç bölümünde özetlenmektedir.

Anahtar Kelimeler: Uçak tasarımı, uçak performansı, açık kaynak yazılım, UPA-Gaziantep

To my **Mother**, who gave me everything

To my **Professor**, who guided me in every step along this research

To my **Wife, Children, and Friends**, who supported me without conditions

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LIST OF SYMBOLS/ABREVIATIONS

α , AoA	Angle of Attack
γ	Path angle
δ	Deflection of control surface
δ_e	Deflection of elevator (in degrees)
δ_t	Deflection of engine throttle (percent)
δ	Pressure ratio P/P_s (Atmosphere)
σ	Density ratio ρ/ρ_s (Atmosphere)
θ	Temperature ratio T/T_s (Atmosphere)
ρ	Air density at flight altitude (Atmosphere)
ρ_s	Air density at sea level (Atmosphere)
η_{prop}	Propeller efficiency
a	Speed of sound
AR	Aspect ratio, from the equation b^2/S
b	Wing span
C_D	Drag coefficient
C_D	Drag coefficient
$C_{D\,i}$	Induced Drag coefficient
$C_{D\,o}$	Parasite Drag coefficient
C_L	Lift coefficient
$C_{L\,max}$	Maximum lift coefficient
C_{Lo}	Lift coefficient at $\alpha = 0$

$C_{L\alpha}$	Lift curve slope
C_m	Aerodynamic moment coefficient
e	Oswald's efficiency factor
E	Endurance
FOSS	Free open source software
g	Gravity acceleration
h	Aircraft Altitude, i.e. Height above sea level
h_{max}	Maximum flight altitude
h_{sc}	Service Ceiling flight altitude
k	Coefficient from Polar Equation ($C_D = \ C_{D_0} + k.C_L^2$)
LRC	Long range cruise
M	Mach number
M_{ne}	Mach number of "Never exceed speed"
M_{mo}	Mach number of "Maximum operating limit speed"
MTOW	Maximum Take-off Weight
n	Load factor
P	Air Pressure (Atmosphere)
P_s	Air Pressure at sea level (Atmosphere)
P	Engine power (usually for piston engines)
R	Range
S	Reference wing area
T	Air temperature (Atmosphere)
T_s	Air temperature at sea level (Atmosphere)
T	Engine Thrust (usually for jet engines)
TSFC	Thrust specific fuel consumption

UPA-Gaziantep	Uçak Performans Analizi - Gaziantep
V	Aircraft airspeed
V_2	Take-off safety speed
V_{app}	Airspeed at threshold in landing approach
V_{ne}	Never exceed speed
V_{mo}	Maximum operating limit speed
V_{stall}	Stall speed
W	Aircraft weight
W_i	Aircraft initial weight
W_f	Aircraft final weight
W_{fuel}	Weight of fuel in the Aircraft

CHAPTER 1

INTRODUCTION

1.1 Research Motivation

In the era of 20th century, sharing information and collaborative projects take the lead of future development. Tesla Motors has removed the Tesla's patents, and all Tesla's patents are now open for the world to use. These efforts contributed to the good of humanity.

Aircraft industry has been limited to big cooperation for long time, but since the revolution of quadcopters; the aircraft design became more practiced on different scales, including students, entrepreneurs, start-ups, and small companies; who are designing and developing new aircraft every day. The design process starts from choosing the main aircraft configurations such as dimensions, weight, wing positioning, airfoils, etc., in addition to the power plant specifications, and landing gear. The aircraft performance is then calculated based on these assumed configurations with each other, the results is the aerodynamic components and limitations of aircraft such as min and max speed, altitude, and static and dynamic stability. Based on these results, the assumed configurations could be accepted, or could be modified and then perform the analysis again. Normally, many configurations suggested and calculated until reaching an optimum design that satisfy the required performance demands. This procedure is very costly and requires time and effort. There are several available solutions with different capabilities, but they are either commercial solutions not affordable by students or written in an outdated programming language and cause many compatibility issues. There are powerful free solutions but they are limited to specific type of aircrafts, others are sophisticated and only specialist can use. Hence, the effort to provide open-source user-friendly aircraft performance software comes in place.

1.2 UPA-Gaziantep Program

In this thesis, designing of open-source easy-to-use aircraft performance program named (Uçak Performans Analizi) *UPA-Gaziantep* is realized. The software is designed in modular structure to enable easy elaboration and development in future, in order to improve the software performance and adapt the software to handle different types of aircrafts such as quadrotors, and unconventional configurations of aircrafts.

The fundamental objective of this program is providing an **easy tool to aircraft designers and students**, in order to aid them in making design decisions that lead to required **performance**.

This thesis aims to encourage more people to design and participate in aircraft industry and open new aspects of creativity.

1.3 Thesis Outline

This thesis is structured into seven chapters. The first chapter contains an introduction to the research. The introduction states the motivation behind this research and present overview of the software developed in this research. The literature review is discussed in the second chapter. It covers a brief description of the aircraft design process and aircraft performance as part of the preliminary design process and the need for computational solutions. To investigate this, an overview of already-existed software is presented. The specification of each software is discussed. The chapter concludes a comparison table that shows the advantages and disadvantages of each software. The comparison identified the gap, where this research focuses. The novelty of this work is discussed at the end of this chapter. The third chapter is dedicated for theoretical study. The aircraft model is derived to be used in the computational iterative solution. The fourth chapter presents the work methodology. First, it introduces both MATLAB programming language and GitHub for open-source software. Then it describes the software development approach. Finally, it explains how to operate the software. The software results are validated by conducting a comparison between program results with already known data for a case study. The fifth chapter presents the case study and presents detailed inputs for the case study to be analyzed via the software. The results and discussion are covered in the sixth chapter. The conclusion discusses the software accuracy, usability, and feathers require further development. The future work is suggested at the end. The thesis also includes several appendices related to the thesis.

CHAPTER 2

LITERATURE SURVEY

2.1 Aircraft Performance in Aircraft Design

Aircraft performance describes the aircraft capabilities and limitation, such as maximum flight altitude (ceiling), maximum speed at specific engine throttle level, maximum range for a given wright, etc. Aircrafts are classified based on their performances such as commercial transport jet, private transport jet, combat jet, military transport aircraft, short takeoff and landing STOL, and civil cargo aircrafts. The aircraft operator selects the aircraft with performance specifications suitable to the required task of the aircraft. While on the other hand, and according to Aircraft Performance Book [1] the aircraft performance specifications are considered inputs for the aircraft design engineer, where the designer seeks to achieve the preferable performance within design limitation, such as weight, manufacturing technology, manufacturing cost and operation cost efficiency, and engine specifications.

Advanced Aircraft Design by Torenbeek [2] describes the aircraft design process. It can be divided into four main parts:

2.1.1 Conceptual Design

Prior to this stage, the market requirements must be decided to validate the need and the feasibility of designing a new aircraft. In this stage, the initial selection of aircraft size and weight, engine type and model, wing shape and position, empennage type. As a result, a 3D model aircraft can be generated with aircraft capabilities and limitations. The aim of this stage is to take decision whether to continue or discontinue the design process.

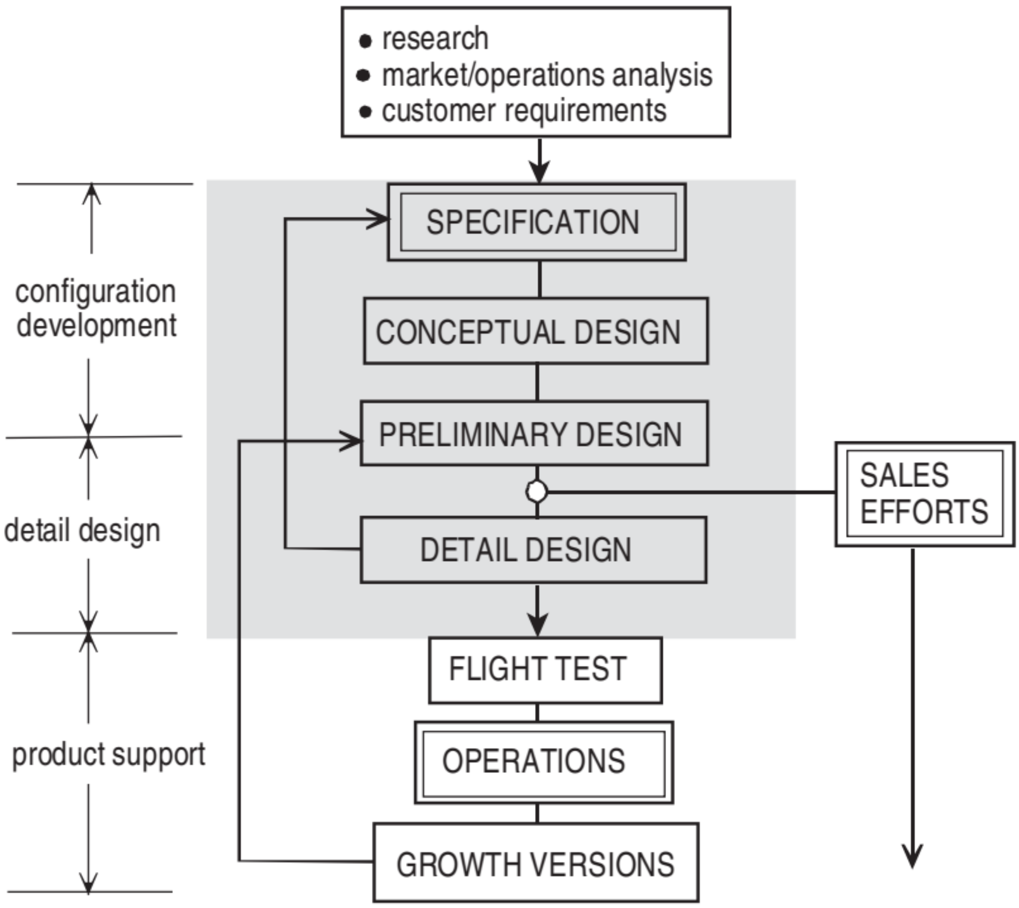

Figure 2.1 Commercial Aircraft Development Process [3]

2.1.2 Preliminary Design

In this stage, core aircraft configurations are set, from which basic calculations are performed to estimate main parameters and generate low-fidelity aircraft model. The work of this research lays mostly within this phase.

2.1.3 Detailed Design

Some references discuss this stage as a separate stage, while others discuss it as a separate one. Both are possible due to the interference between the two phases. In this stage, detailed geometry of aircraft parts and components are specified. Then, the production processes is planned to identify the possibility and cost of manufacturing the aircraft. The aim of this stage is to get high fidelity design plans and diagrams. During this stage, some challenges in the design may force the designer to loop with previous stage back and forth several times.

2.1.4 Manufacturing and Testing

The results from the previous stages are turned into reality. Some components maybe re-designed based on deviation between theoretical designed parts and actual results from manufactured parts. Therefore, this stage also overlap with the previous stage.

The diagram below as shown on Civil Jet Aircraft Design [4] illustrates the aircraft designing and manufacturing flowchart, showing that phases are not independent phases, but they have overlapping due the effect of each phase on the results on the next phase. Cost increases exponentially as the design stage progresses. This is very important factor for designers to spend more time in preliminary design phase, due to the low cost relatively of this phase.

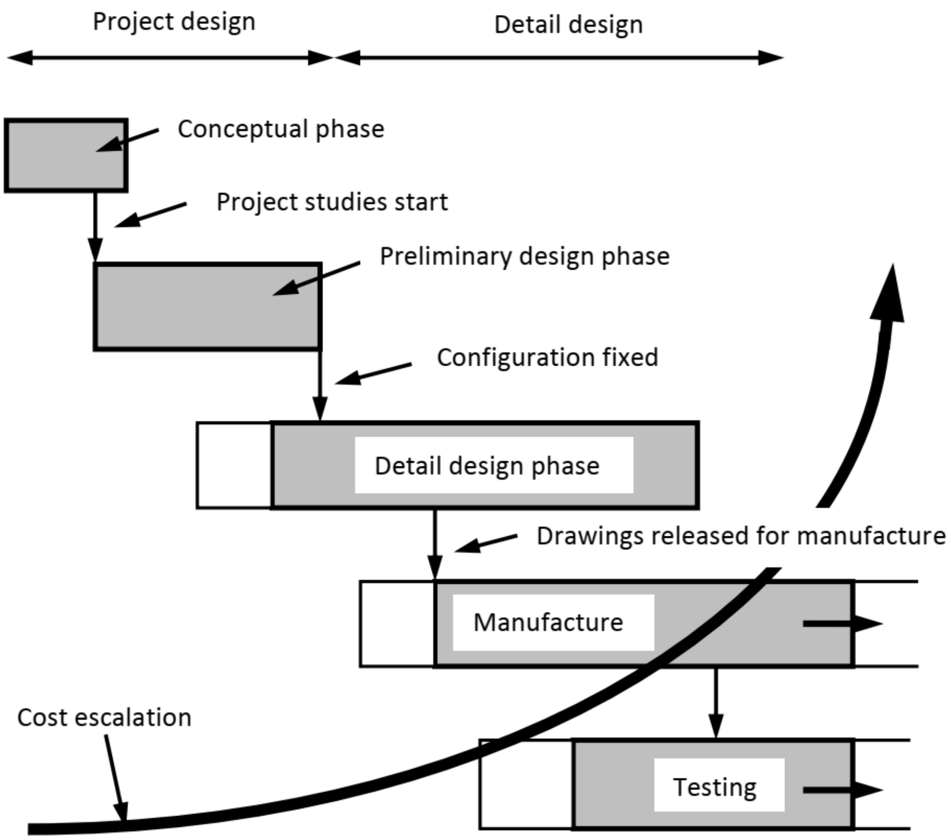

Figure 2.2 Design and Manufacturing Schedule [4]

2.2 Preliminary Design in Aircraft Design

One good presentation of the aircraft preliminary design is the flowchart shown in Figure 2.3 presented in the book Advanced Aircraft Design by Torenbeek [2]. The aircraft performance block lays near to the end of this process. The results of this block need to match the design requirement, if not the designer needs to re-iterate the whole process.

The preliminary design steps have several approaches. The design approach described by D. Raymer [5] has been quoted in many references. There are three steps proceed the aircraft performance estimation step. Here is a brief description of those steps.

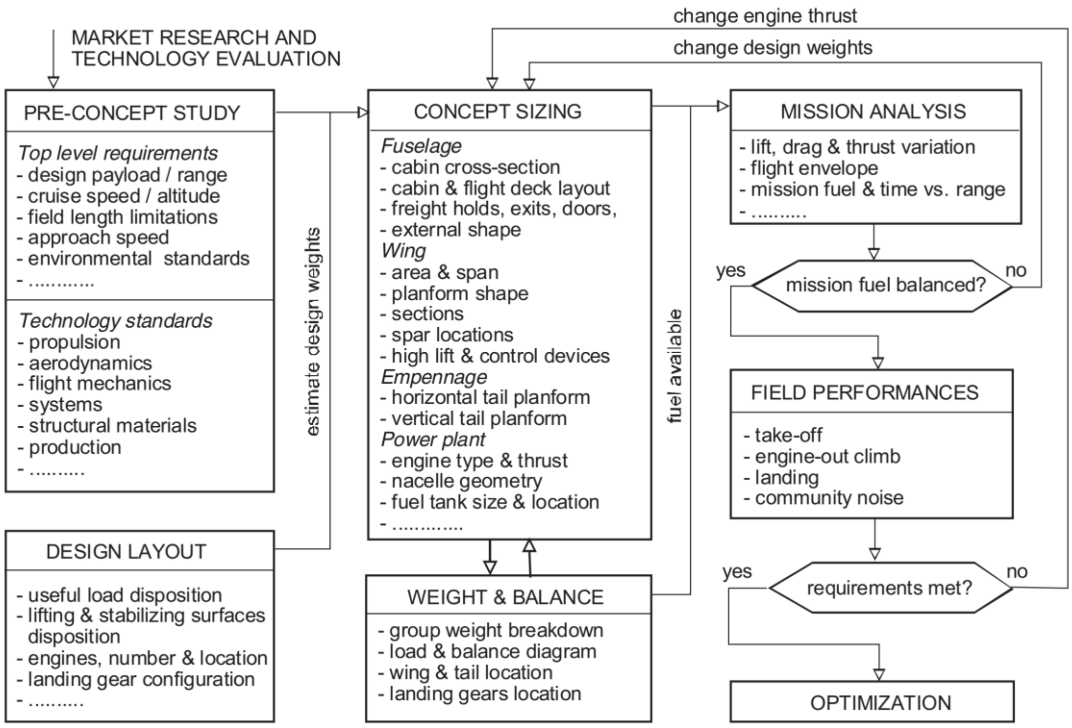

Figure 2.3 Jet Preliminary (Baseline) Design Process [2]

2.2.1 Aircraft Class

Aircrafts can be classified in different categories. According to the type of engine; classification can be propeller aircraft, jet aircraft or even a glider for non-powered aircraft. Also, according to purpose of the aircraft, the classification can be commercial transport jet, commercial cargo jet, private businesses jet, fighter jet, or even single seat aircraft. From configurations prospective, the classification is conventional aircraft with fixed low wings for, or aircraft with canard. There are other classifications as well.

The basic requirement determines the aircraft class. Aircraft classes can help the designer to select the basic aircraft configurations. The selection of less conventional aircrafts impose more challenges during design process due to lack of available data and experience. The advanced new material and structural technologies adopted within aircraft industries offer more variety of options for today engineers and designers. Nevertheless, only general selection is required at this step. The structure optimization is performed during the detailed design phase.

2.2.2 Flight Systems and Power Plant

In this step, the basic flight conditions are set, including maximum speed and Mach number, maximum altitude. Those parameters are linked to the power plant selection. There are several types of engines including turbofan, turboprop, turbojet, or piston engine with propeller. The engine relates not only to the flight parameters, but also to the aircraft configurations.

2.2.3 Body and Wings

Aerodynamics of aircraft determines lift and drag, thus aircraft payload and maneuverability. The aerodynamics are determined mainly from aircraft body, wings and tail. The fuselage selection define aircraft dimensions, loading capacity, either passengers or cargo, and fuel capacity. Wings selection is essential to aircraft performance. The selection includes wing airfoil, geometry, and space of lift surface.

2.2.4 Aerodynamics and Mass

From aircraft wings airfoil and area, aircraft aerodynamics can be estimated. The aircraft initial mass can be estimated from aerodynamics and power plant specifications. In this stage, simple analytical calculations are used to obtain acceptable results. The later steps require detailed numerical calculations to predict a more accurate aerodynamics including the wing-fuselage interaction.

2.2.5 Performance Analysis

The initial aerodynamics and power plant initial calculation obtained in previous steps are the inputs to estimate aircraft performance. In this step, the designer can analyze the aircraft performance that is represented by range and endurance with payload, cruise performance, maximum flight speed and altitude, climb and decent performances.

2.3 Aircraft Performance Estimation

Performance calculations are derived and carried out at main phases during flight. The main phases in the flight path are Cruise, Take-off, Climb, and Landing. The main performance design analyses for aircrafts is carried out for cruise, nevertheless; take-off and landing parameters are very important for airlines and aircraft potential buyers.

Aircraft performance can be estimated in two methods:

The first method is to carry out physical simulations using full-size or scaled-size detailed models during the conceptual design stage. Although this method has high accuracy, but it requires a lot of time and money.

The second method suggests using statistical data and existing aircrafts information from successful aircrafts. Then put these data into mathematical equations to predict performance parameters. The mathematical equations have been derived and presented in different references by different authors. The advantage of this methodology is it can be carried out very fast at a low cost, while the results higher error margin.

Disregarding to the design method, the performance analysis is a complicated process with reputation nature. Thus, many efforts were made to computerize the process. Various computer and programs software are already available. The following paragraph shed light on the programs available.

2.4 Existing Performance Analysis Software

2.4.1 Introduction

As has been discussed before, the performance calculations are very vital for anyone involved in aircraft design, including designers, engineers, students, and hobbyists. Therefore, many efforts have already been made to computerize the process and offer practical tools to be used.

The software/code developed for the aircraft performance can be categorized into two main groups:

The first group is the professional comprehensive software packages. Those programs are usually commercial programs targeting aircraft manufacturing. Good examples are the Aircraft Performance Program (APP) [6], CEASIOM [7].

The other group is patches of codes developed to solve one equation or more during the analysis process. One example is Empennage Sizing and Aircraft Stability using MATLAB [8]. In this project, the researcher created MATLAB code in order to solve a very specific problem: how to properly size the empennage of a low speed aircraft for a desired level of stability. Many other examples can be found in the aircraft design books.

The first group packages share common character, that they require strong prior knowledge from the user in aircraft design, since they have been designed for commercial purposes, and they are not intended for educational use. The license for these software vary from 99 € to over a thousand euros. The cost could impose restriction of access for students and innovators. On the other hand, the majority of second group are codes written using FORTRAN. Despite the fact, FORTAN is still quite used among researchers, yet many students prefer to use more flexible programming languages such as MATLAB and Python.

It is worth mentioning that there is wide range of studies and software for aircraft performance related to Air Traffic Management (ATM) decision support, which is another category of performance that lays outside of this research scope and it has not been discussed. For example; the Ph.D. thesis by (Tolga, 2010) [9] has developed an aircraft performance model for ATM. Tolga work included deriving model with optimization algorithms to predict aircraft trajectory.

To have a better understanding of available solutions, a comparison is presented in the following section. The comparison highlight the specifications and limitation of each software. It is important to highlight that the comparison studies programs dedicated for performance estimation during the conceptual design process, while performance fro ATM software are not included.

2.4.2 Aircraft Design Software (ADS)

Figure 2.4 ODA Website Header

Aircraft Design Software (ADS) [10] is conceptual design software for light aircrafts. It is presented on the market commercially. There are five versions of the software with different access to modules with price different. Although the software is dedicated for aircraft design, it has some crucial performance parameters as an output from the computation. The software is by Optimal Aircraft Design (OAD) [11]

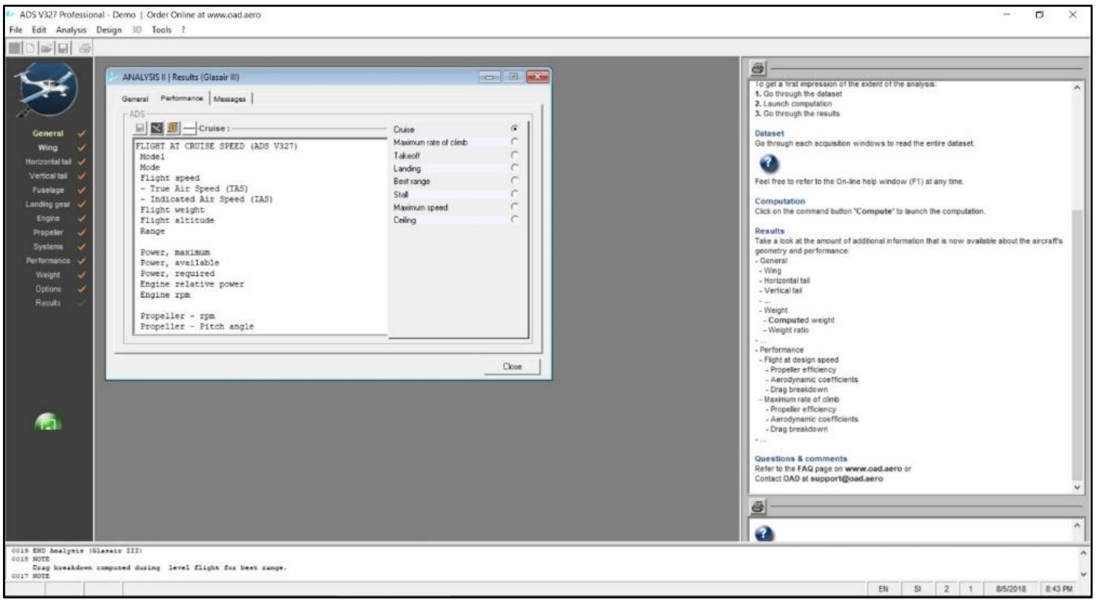

Figure 2.5 Screenshot form ADS Software [10]

<u>User Expertise:</u> The Demo version of the software was downloaded and tested. It can found from the test that the software is suitable for users who have medium to advance experienced in aircraft engineering and aircraft design, to be able to fill in

all inputs of each model to be designed.

Aircraft Performance Results: The program has several modules. The performance module calculate the following: Cruise, Maximum rate of climb, Takeoff, Landing, Best Range, Stall, Maximum Speed, and Ceiling. Results are viewed as text.

Development possibility: The software can calculate several type of light aircraft, but it cannot be editable or modified.

Cost: Educational ADS 995.00 € + 20% for annual maintenance

Operating environment: The software is a standalone windows program.

2.4.3 Design Airspeeds and Flight Envelope Calculator

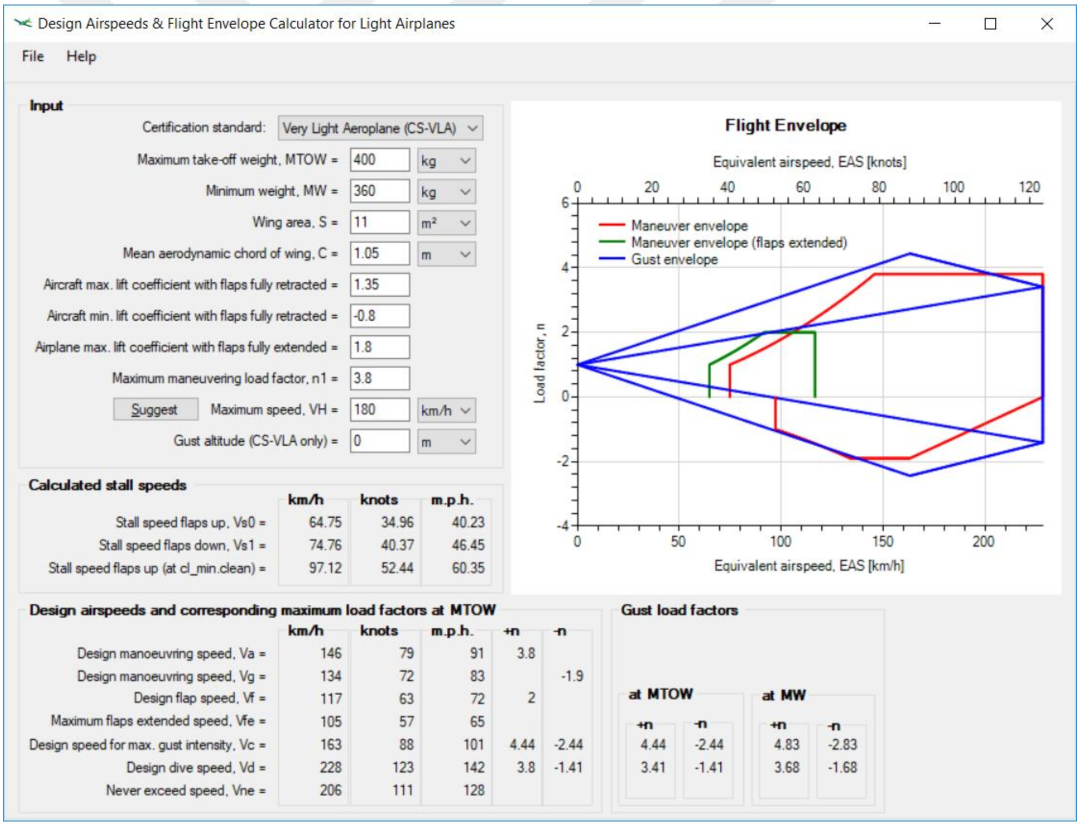

Figure 2.6 Screenshot from Design Airspeeds and Flight Envelope Calculator

Design Airspeeds & Flight Envelope (V-n diagram) Calculator for Light Airplanes [12], as the title indicates, is a compact application for quick calculation and drawing of flight envelope (V-n diagram) for a given airplane configuration of a light airplane.

User Expertise: The software is easy to use. The user can select the certification standard, then insert basic aircraft specifications: Maximum take-off weight (MTOW), Minimum weight (MW), Wing area (S), Mean aerodynamic chord of wing (C), Maximum maneuvering load factor (n1), and Aircraft max. & min lift coefficient with flaps fully retracted and extended.

Aircraft Performance Results: The software has one output, which is V-n diagram, which is one of aircraft performance essential diagrams.

Usage limitation: The software is suitable only for ultralight aircraft, light sport airplanes and very light airplanes. The software is not designed to be used as educational tool for aeronautical engineering students.

Development possibility: The software can calculate several type of light aircraft, but it cannot be editable or modified.

Cost: There are free version available for download. The full version is 99.00 €.

Operating environment: The software is a standalone windows program.

2.4.4 Aircraft Performance Program (APP)

Aircraft Performance Program APP [6] is an advanced aircraft performance software was produced by by ALR Aerospace, in association with RUAG Aerospace Defense Technology. It covers both; the conceptual and preliminary aircraft design.

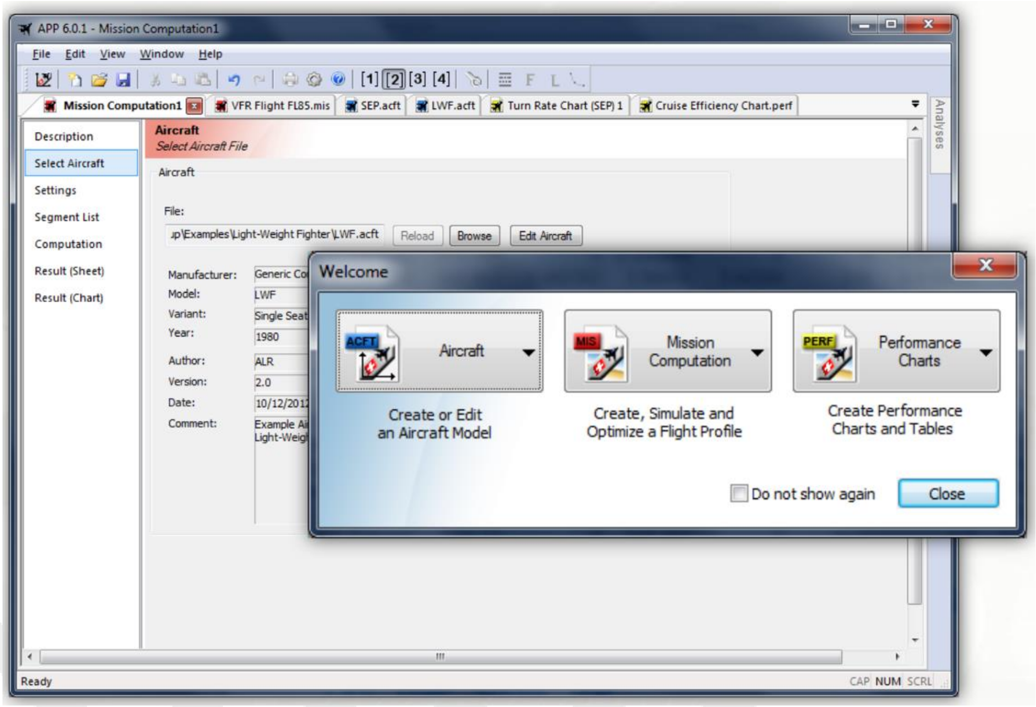

Figure 2.7 Screenshot from AAP Interface

<u>User Expertise:</u> The software is designed for advanced users; aircraft design engineers with experience about aircraft design, and it requires well-defined configurations and specifications for the aircraft to be designed.

<u>Aircraft Performance Results:</u> The software calculates point performance and mission performance. The software have rich outputs of performance diagrams and more than 60 parameters.

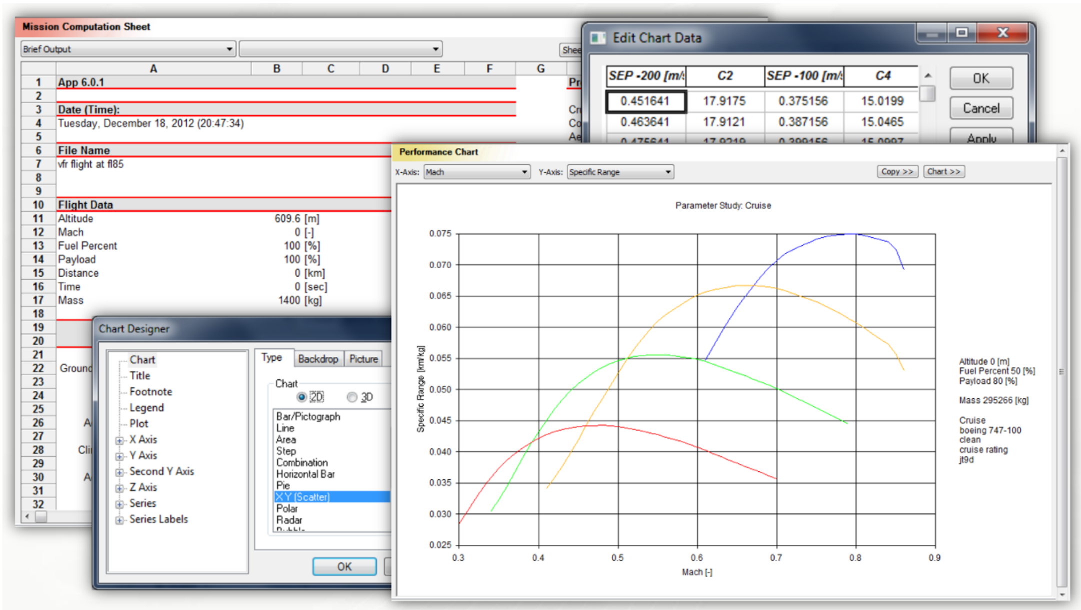

Figure 2.8 Screenshot from AAP Results View

Development possibility: The software can calculate several type of light aircraft, but it cannot be editable or modified.

Cost: The software has quotation request form, with different software license pricing: Commercial, University, Small Business (< 20 employees), and Individual/Hobbyist (NFP).

Operating environment: The software is a standalone windows program.

2.4.5 CEASIOM

Computerised Environment for Aircraft Synthesis and Integrated Optimisation Methods (CEASIOM) [7] is a software developed by SimSAC partners as a Conceptual Aircraft design tool, with multi modules, each module solve part of the design process. The software can be considered one of the most comprehensive design software available. CEASIOM have been used for Boeing 747-100 analysis by Richardson [13].

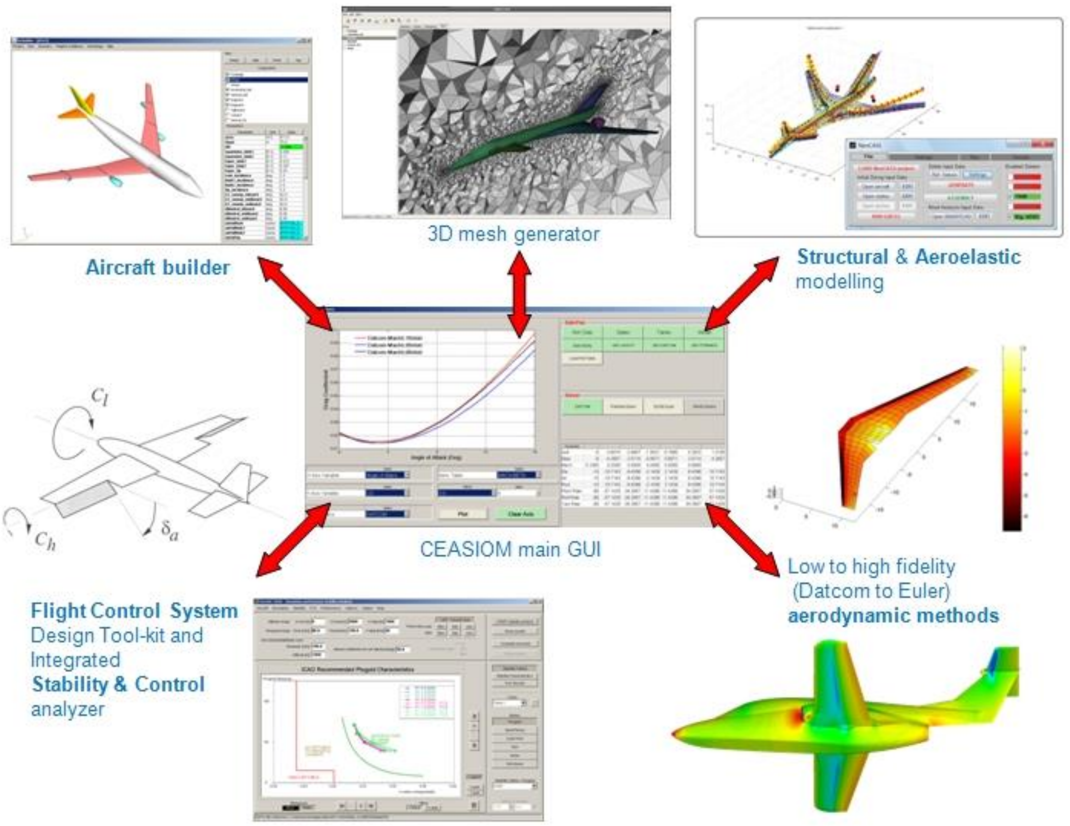

Figure 2.9 CEASIOM Modules [14]

User Expertise: CEASIOM user experience vary from module to another. Nevertheless; generally, the software can be considered a software for advanced users; such as aircraft design engineers. Results requires many inputs into the various design modules.

Aircraft Performance Results: The module SDSA (Simulation and Dynamic Stability Analyser) covers different design outputs including the "Performance prediction". The results can be obtained as text results or MATLAB plots.

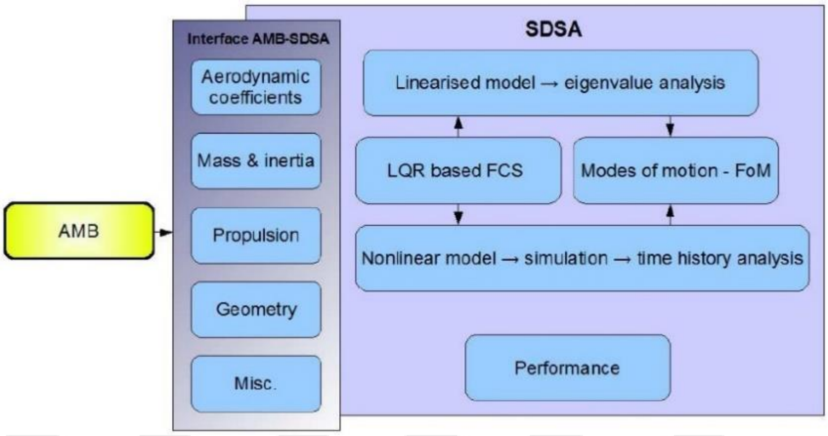

Figure 2.10 Structure and Functionality of SDSA [15]

Usage limitation: Despite the fact that the software is available free for download on the software website, but the current version has compatibility issues with the MATLAB versions available during this research (MATLAB R2017a). This led to limitation in using or even testing of the software.

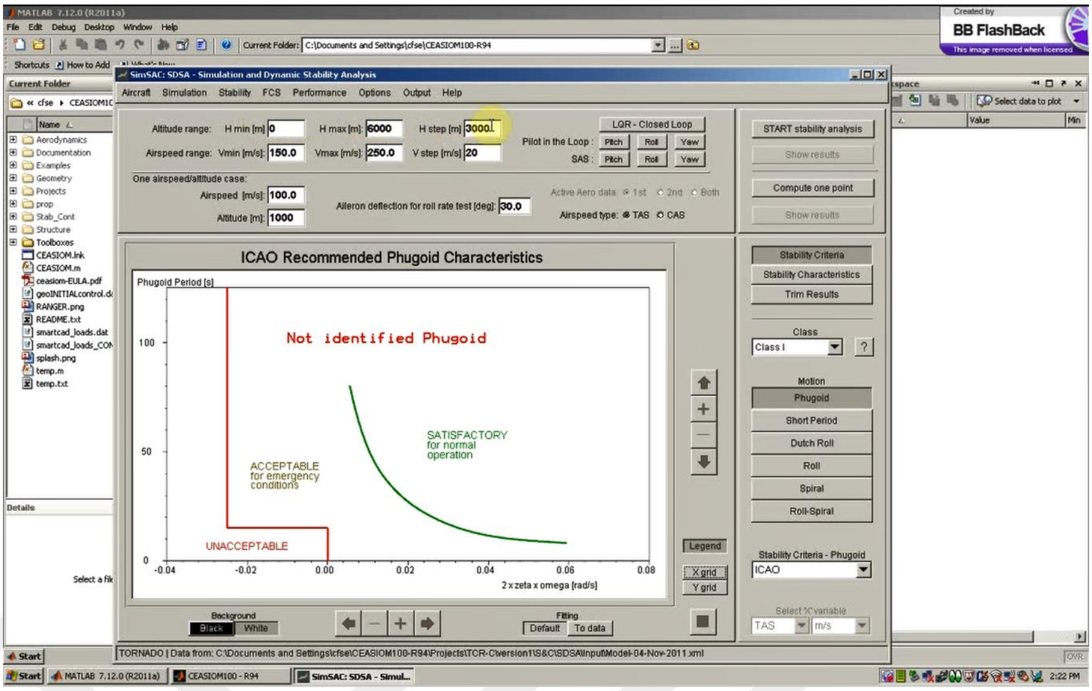

Figure 2.11 Snapshot from Tutorial Video about CEASIOM [16]

<u>Development possibility:</u> The current MATLAB code are in protected format (Pcode) which are heavily obfuscated format that can be obtained from the m-files. For that the current code cannot be developed or debugged.

<u>Cost:</u> The CEASIOM software is freeware. It is free to download, and user must agree the EULA before downloading.

<u>Operating environment:</u> CEASIOM is MATLAB-based and run within the MATLAB environment. Currently a Python based version of CEASOM named “CEASIOMpy” is under development to replace the current version and solve the compatibility issues.

2.4.6 Flight Performance Software (FLIGHT)

Flight Performance Software (FLIGHT) [17] is used for the prediction and modelling of fixed wing aircraft performance. The software was developed by Dr. Antonio Filippone, aerospace researcher, author, and lecturer.

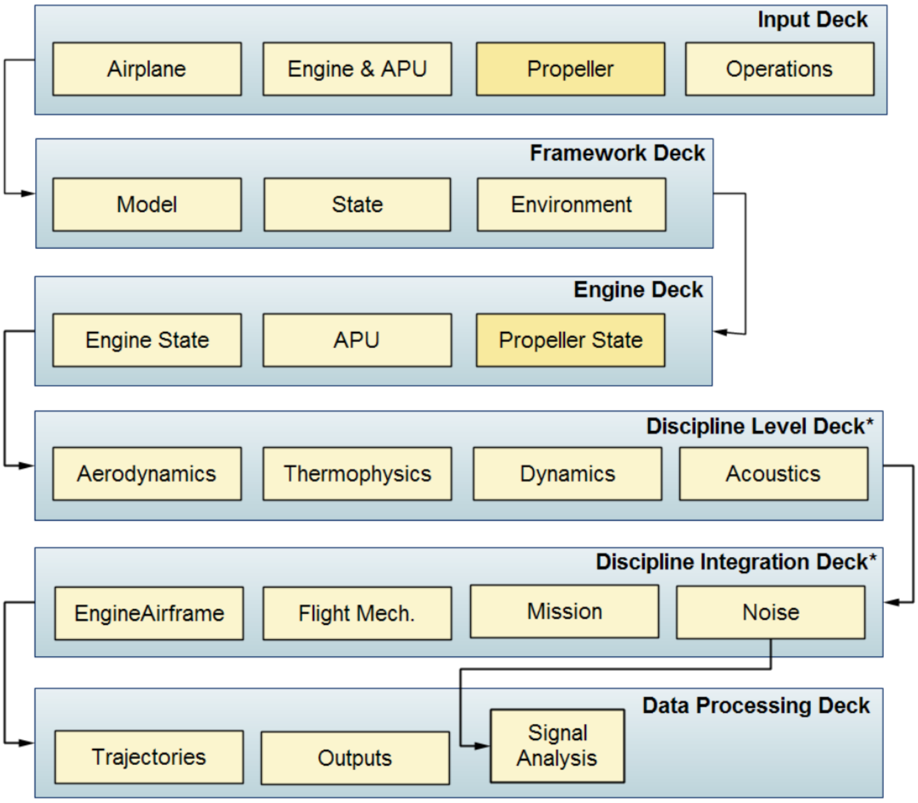

Figure 2.12 Disciplines Modelled in FLIGHT Program [18]

User Expertise: Based on the software manual [19], the software focuses on advanced users with detailed aircraft information. The software runs via "command line" and has no user-interface.

Aircraft Performance Results: The software result menu has dedicated menu for Performance chart. The results options are viewed in Figure 2.13.

```
Performance Charts
____________________________________

Aerodynamics              [1]
Specific Air Range        [2]
Engine Envelopes          [3]
Flight Envelopes          [4]

Propeller                 [5]
WAT (AEO take-off)        [6]
Balanced Field Length     [7]
Payload-Range             [8]
Economic Mach no.         [9]
CG Effects               [10]
Buffet Boundary          [11]
Spec. Excess Power       [12]
Go-Around Charts         [13]
Atm-Speed Charts         [14]
Holding Speeds           [15]
Max Descent Rates        [16]

V-n diagram              [17]
Climb Polar              [18]
Min. Control Speed       [19]
Gust Response            [20]
Longitudinal dynamics    [21]
Aircraft volumes         [22]

Upper level [any key]
```

Figure 2.13 Performance Charts Options [19]

Below is an example of output report for Cruise Performance of an Airbus A320 Airplane Model, as presented by software manual [19].

```
--------------------------------------------------------
FLIGHT Version : 7.1.7
Revision : b
Database : 20.2.0
Prop_Noise : 3.8.1
Build : 4846/50.7%
Licensed to : Owner
--------------------------------------------------------
JOB IDENTITY
--------------------------------------------------------
Run Time: Monday 24 November 2014 at 16:53
Computer platform is Windows
Airplane/Engine/Data are CLASSIFIED
Airplane = Airbus A320-200-CFM; Version 1.3.0
Engine = CFM56-5C4P ; Version 3.1.1
APU = 131-9
--------------------------------------------------------
[Segment]: Cruise: X = 557.8 [nm]; FL-380; ICW = 61.746 [ton]; M = 0.748
h FL X time fuel fflow vc vc
[m] [n-m] [min] [kg] [kg/s] [m/s] [f/min]
----------------------------------------------------------------------------
-
11582 380 557.8 77.23 2460.7 0.531
----------------------------------------------------------------------------
-
Summary: 557.8 77.23 2460.7 0.531
```

Usage limitation: The software is designed for transport airplanes powered by turbofans and turboprops engines. Demos of an older version are available for free download [20]. The demo versions have expired in 2014, so they cannot be tested now.

Development possibility: the code is not available for personalization or development.

Cost: The FLIGHT software is not free. License can be obtained from University of Manchester Intellectual Property Services.

Operating environment: The software is programmed in FORTRAN. For that, the program runs under MS-DOS command line, and has Linux version as well.

2.4.7 Comparison between Available Software

To summarize the information mentioned in the reviews above, a comparison table for is prepared. In the table, only main aspects are mentioned. The UPA-Gaziantep software, which is developed during this research, has been added as well.

Table 2.1 Aircraft Performance Software Comparison

	User expertise	*Usage limitations*	*Development possibility*	*Cost*	*Operating Environment*
Aircraft Design Software (ADS) [11]	Medium to Advanced	-	Not Available	Educational ADS 995.00 €	Windows standalone software
Design Airspeeds and Flight Envelope Calculator [12]	Basic	Specific type of aircrafts	Not Available	full version is 99.00 €	Windows standalone software
Aircraft Performance Program (APP) [6]	Advanced	-	Not Available	Request for Quotation	Windows standalone software
CEASIOM [7]	Advanced	Outdated	Not Available	Free	MATLAB-based code
FLIGHT [17]	Advanced	Specific type of aircrafts	Not Available	Request for Quotation	MS-DOS Command line
UPA-Gaziantep	University undergraduate and graduate Students for better understanding of aircraft design	Require more development	Open-source and modular based	Free	MATLAB-based code

2.5 Novelty Approach

The review for existing work shows that there is space for a software dedicated for undergraduates and postgraduates aeronautical engineering students. To make an educational software, the software needs have essential characteristics:

2.5.1 User Friendly (Easy to Use)

Students usually do have advance experience, and need simple tools to support the learning process. To make UPA-Gaziantep user friendly, it must be simple and easy to use. To do so, two things were taken in consideration: A) Using Graphical User Interface (GUI) to allow better user experience. MATLAB GUI was used in the development of UPA-Gaziantep. B) Minimum inputs required to perform meaningful calculation and produce results. The challenge can be overcome from two aspects: First, adopting equations that require minimum inputs, which result in less accurate results, as trade-off for ease-of-use. Second, allow more inputs from external data, such as aerodynamic characteristics of airfoil or jet engine specifications.

2.5.2 Instructive (Educational)

To enable the program to be used an instructive tool by aerospace engineering students, the calculations need to be possible to understand. This can be done by: A) Insert detailed comments in the code to explain the steps being done at each part. B) Using readable variable names. For example V_max is easy to read as V_{max}.

2.5.3 Open Source (Possible to Test/Adapt/Personalize)

The code needs to be possible to change, so users, including students, can change line to test the implications on the results, or add new module to the code to solve specific case they have. UPA-Gaziantep is made an free open-source software (FOSS), that complies with criteria of Open-source Initiative (OSI) as per mentioned by [21]. The software structure is made easy to understand, edit, and develop by adding new modules.

CHAPTER 3

THEORETICAL STUDY

3.1 Flight Profile

A general aircraft normally operates within three main flight phases, which are 1) Take-off and climbing, 2) Cruising, and 3) descent and landing (Figure 3.1). In addition, an airplane could glide as the case of the gliders or sailplanes, or in case of power-off flight. Usually in the performance analysis, the steady (un-accelerated) flight is generally taken into account.

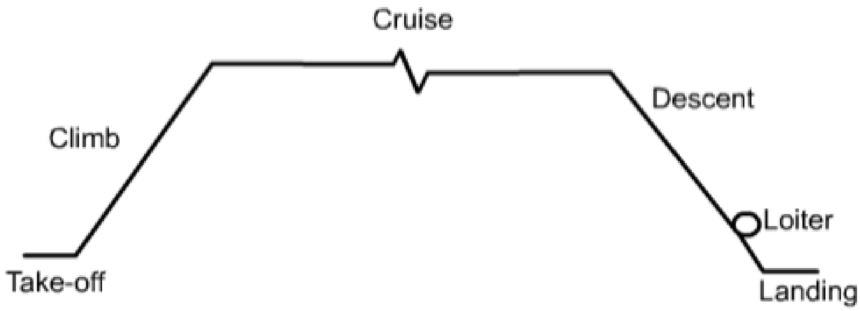

Figure 3.1 Typical Flight Path of a Passenger Airplane

Two main types of motions are considered when analyzing the aircraft performance:

1) Un-accelerated flights

    a. Steady level flight

    b. Climb, glide and descent

2) Accelerated flights

    a. Accelerated level flight and climb

    b. Turn, loop, and other maneuvers along curved paths

c. Take-off and landing.

The program developed in this thesis focuses only on the performance calculations for the steady (un-accelerated) flights. While it can be extended in the future to include other flight phases.

3.2 Forces on Aircraft

To analyze an aircraft motion, it is studied as a rigid body that has many types of forces acting on it during each phase of the flight [22]. These forces are:

- Gravitational force (Aircraft Weight),
- Aerodynamic forces (Lift and Drag),
- Propulsive force (Thrust), this force do not exist in gliding flight.

In the following sections, a brief about each force is presented.

3.2.1 Lift

Lift Force is the vertical component of the aerodynamic resultant force, which acts on the aerodynamic center. It is perpendicular on the velocity direction and opposites the downward force of the weight. The wing produces the major fraction of the aircraft lift. The shape of the airfoil creates a difference in pressure distribution between the upper and lower surfaces of the wing. This difference is what produces the lift. At the same time, this pressure difference also varies with the relative speed and direction between the wing and airflow direction.

Equation (3.1) is the mathematical relationship of lift. It shows that the lift is related to the angle of attack (AoA), airspeed, altitude, and the area of the wing [23].

$$L = \frac{1}{2}\,\rho\,V^2\,S\,C_L \tag{3.1}$$

Where, ρ is air density, V is the airflow speed, S is the reference wing area, and C_L is the dimensionless lift coefficient. CL has a strong dependence on the aircraft angle of attack α according to:

$$C_L = C_{Lo} + C_{L\alpha}.\alpha \tag{3.2}$$

Where C_{Lo} is the lift coefficient at $\alpha = 0$, and $C_{L\alpha}$ is the lift curve slope.

At high angle of attack the air flow may become turbulent causing the profile to "fail" or "stall". This limits the maximum possible lift coefficient $C_L \leq C_{L\,max}$; where $C_{L\,max}$ is when α reaches α_{stall} the angle of attack before stall.

3.2.2 Drag

Drag Force is the horizontal component of the aerodynamic resultant force, also acts on the aerodynamic center and parallel to the velocity direction. Drag minimization is a major concern for designers. Two different types of drag on an aircraft, parasite and induced.

Parasite Drag: This type is composed of three types as following:

- Form Drag, which is air resistance to motion due to the shape of the aircraft,
- Skin Friction Drag, which is due to the smoothness or roughness of the aircraft surfaces, and
- Interference Drag, which is due to the interaction between surfaces with different characteristics (e.g. wing and fuselage).

The wing of alone has very low parasite drag, but when the total drag of the aircraft is added to it, the amount of drag becomes significant. Parasite drag can be reduced by streamlining the non-lifting parts of the sailplane, like fuselage, and keeping all aircraft surfaces as smooth as possible.

Induced Drag: this part of drag arises from the development of lift and it is typically written as [23]:

$$C_{Di} = \frac{C_L^2}{\pi \, AR \, e} \tag{3.3}$$

Where, AR is the aspect ratio, e is Oswald's efficiency factor. As equation (3-2) indicates, the larger the aspect ratio of the wing is, the lesser the induced drag is. Therefore, aircraft designers always aim to reduce drag by increasing the aspect ratio of the glider.

The total drag is written mathematically as following [23] [24]:

$$C_D = C_{Do} + C_{Di} \tag{3.4}$$

Where, the first term is the form drag and the second is the induced drag.

3.2.3 Weight

The weight is the gravitational force that opposes the lift. This force acts on the center of gravity of the aircraft and it is given by:

$$W = m * g \tag{3.5}$$

Where W is the aircraft weight, m is the aircraft mass, and g is the gravity acceleration.

Weight is a key factor in performance analysis, as it influences thrust (power required). Therefore, it is common objective to reduce the weight of aircraft as much as possible.

3.2.4 Thrust

The thrust is the propulsive force that opposes the drag and make the aircraft moving forward. The thrust principle is explained according to Newton's third Law, where the engines push air back with the same force that the air moves the plane forward. The thrust could be produced from two main engines, the jet and propeller engines. In gliding flight, the gravity is what provides thrust for the aircraft. At steady level flight, the thrust equals drag.

3.3 Aircraft Performance Analysis

This section contains the equation required for performance calculations. The equations have been concluded and explained in a good number of references [23] [24]. Therefore, in this section, the equations necessary for program developed are only listed.

3.3.1 Aircraft Steady Gliding Flight

At the gliding flight, the aircraft is subjected only to three forces, which are lift, drag, and weight (Figure 3.2). In other words, there is no a propulsive force.

$$-D + W \sin \gamma_{glide} = 0\,, \tag{3.6}$$

$$L + W \cos \gamma_{glide} = 0 \tag{3.7}$$

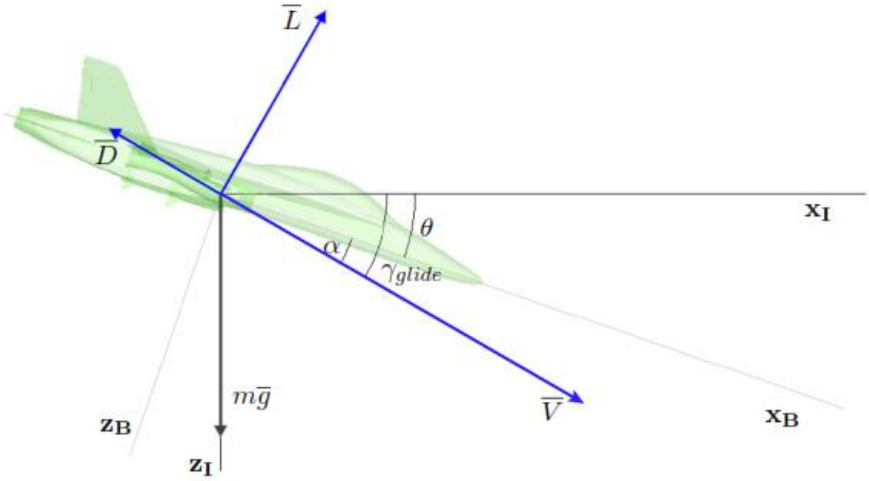

Figure 3.2 Steady Gliding Flight [24]

For defined aircraft velocity V and altitude h, the glide angle γ_{glide} for steady gliding flight can be calculated from the following equation (where ρ is related to h):

$$\gamma_{glide} = \frac{\frac{1}{2}\rho\ V^2\ S\ C_{D0}}{W} + \frac{K\ W}{\frac{1}{2}\rho\ V^2\ S} \tag{3.8}$$

The minimum glide angle $\gamma_{glide\ min}$ occurs at the condition of maximum aerodynamic efficiency i.e. at maximum lift to drag. The minimum glide angle is given by:

$$\gamma_{glide,min} = 2\sqrt{K\ CD_0} \tag{3.9}$$

By satisfying the condition of the lift equals to weight, the air speed for a minimum glide angle $V_{glide\ min}$ is given as following:

$$V_{glide\ at\ \gamma_{min}} = \sqrt{\frac{2\ W}{\rho\ S}\ \sqrt{\frac{K}{CD_0}}} \tag{3.10}$$

From equations mentioned above, the flight velocity can be calculated for defined flight path angle γ_{glide}, as following:

$$\frac{1}{2}\frac{\rho\, S\, C_{Do}}{W}\, V^4 - \gamma_{glide} V^2 + \frac{2\, K\, W}{\rho\, S} = 0 \tag{3.11}$$

This equation can be solved by MATLAB, and two values of V are obtained, V_{max} and V_{min}. by plotting V_{max} and V_{min} at various altitude (h), the flight envelop is plotted.

3.3.2 Aircraft Cruise in Steady Level Flight

At this flight, four forces are acting on the aircraft (Figure 3.3). Steady level flight implies zero acceleration of the aircraft and, thus, the sum of the force vectors on the aircraft is zero.

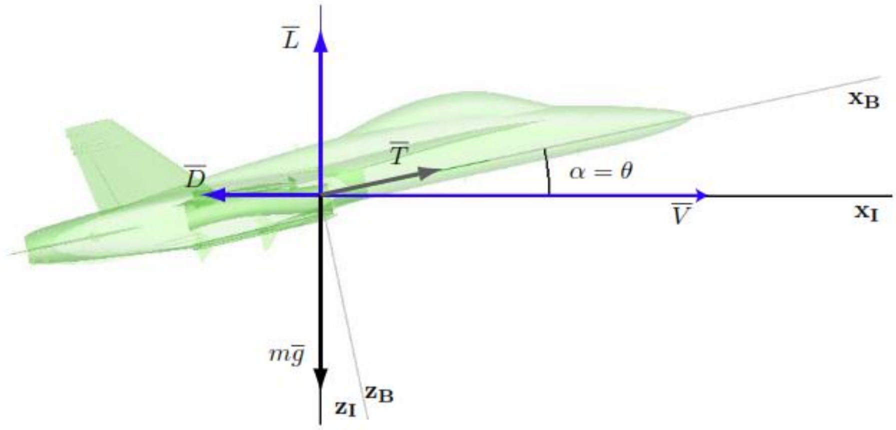

Figure 3.3 Steady Level Flight [24]

The stall constraint for steady flight is expressed as following:

$$C_L \leq C_{L\,max} \qquad \Leftrightarrow \qquad \alpha \leq \alpha_{stall}, \tag{3.12}$$

$$V_{stall} \leq V \tag{3.13}$$

The stall angle of attack (α_{stall}) and stall velocity (V_{stall}) are given by these equations respectively:

$$\alpha_{stall} = \frac{C_{L\,max} - C_{L\,0}}{CL_{\alpha}} \tag{3.14}$$

$$V_{stall} = \sqrt{\frac{2\,W}{\rho\; S\; C_{L\,max}}} \tag{3.15}$$

Where, $C_{L\,max}$ is the maximum lift coefficient, $C_{L\,o}$ is the lift coefficient at zero angle of attack, ρ is the air density at the flight altitude, S is the wing reference area, and W is the aircraft weight.

The second constraint is the thrust/power produced by the engine. The conditions for maximum thrust can be calculated in jet-powered aircraft of propeller-powered aircraft differently. Following sections contain equations for each case.

3.3.2.1 Jet Aircraft Steady Level Performance

Maximum and minimum flight speeds: At full throttle, the thrust required for steady level flight is equal to the maximum thrust that the engine can provide; this leads to the equation:

$$\frac{1}{2}\,\rho\,V^2\,S\,CD_0 + \frac{2\,K\,W^2}{\rho\,V^2\,S} = T^s_{max}\,(\frac{\rho}{\rho^s})^m \tag{3.16}$$

To enable numerical solution for previous equation, it can be re-written as following:

$$\frac{1}{2}\,\rho\,S\,CD_0\,.\,V^4 - T^s_{max}\left(\frac{\rho}{\rho^s}\right)^m .\,V^2 + \frac{2\,K\,W^2}{\rho\,S} = 0 \tag{3.17}$$

This equation typically has two positive solutions. The solution with high value is the maximum possible air speed of the aircraft; the solution of the low value is the minimum possible air speed of the aircraft.

The thrust required to maintain steady level flight is obtained as following:

$$T = \frac{1}{2}\,\rho\,V^2\,S\,CD_0 + \frac{2\,K\,W^2}{\rho\,V^2\,S} \tag{3.18}$$

Where, V is the airspeed, C_{Do} is the drag coefficient at zero angle of attack, K is factor between the aspect ratio (AR) and Oswald efficiency factor (e) given by K = $\frac{1}{\pi\,e\,AR}$.

A jet engine can provide a maximum rated thrust; and the thrust required for steady level flight must not exceed this maximum thrust. In mathematical team, the thrust T required for steady level flight must satisfy:

$$T \;\leq \delta_t \, T^s_{max} \, (\frac{\rho}{\rho^s})^m \tag{3.19}$$

Where, T is the required thrust, δ_t is the throttle setting, T^s_{max} is the maximum thrust at sea level, ρ is the air density at the flight altitude, ρ^s is the air density at sea level.

The constraint on the throttle setting express the constraint of the jet engine thrust. Mathematically is given by:

$$\delta_t = \frac{T}{T^s_{max}} \left(\frac{\rho^s}{\rho}\right)^m \;\leq 1.0 \tag{3.20}$$

The minimum value of the thrust in jet aircrafts, which corresponds the minimum drag, required for steady level flight is given by:

$$T_{min} = 2W \sqrt{K \, C_{Do}} \tag{3.21}$$

3.3.2.2 Propeller Driven Aircraft Steady Level Performance

Maximum and minimum flight speeds: At full throttle, the power required for steady level flight is equal to the maximum power that the engine can provide; this leads to the equation:

$$\frac{1}{2} \, \rho \, V^3 \, S \, CD_0 + \frac{2 \, K \, W^2}{\rho \, V \, S} = \; \eta \, P^s_{max} \left(\frac{\rho}{\rho^s}\right)^m \tag{3.22}$$

To enable numerical solution for previous equation, it can be re-written as following:

$$\frac{1}{2} \, \rho \, S \, C_{Do} \, .V^4 - \eta \, P^s_{max} \left(\frac{\rho}{\rho^s}\right)^m .V + \frac{2 \, K \, W^2}{\rho \;\; S} = \; 0 \tag{3.23}$$

Where, P is the required power, P^s_{max} is the maximum power at sea level, η is propeller efficiency, ρ is the air density at the flight altitude, ρ^s is the air density at sea level.

By solving equation numerically or graphically to obtain two solutions. The solution with higher value is the maximum possible air speed of the aircraft; the solution lower value is the minimum possible air speed of the aircraft.

The thrust required to maintain steady level flight is obtained as following:

$$P = \frac{1}{2}\rho V^3 S\, C_{Do} + \frac{2\,K\,W^2}{\rho\, V\, S} \tag{3.24}$$

Where, V is the airspeed, C_{Do} is the drag coefficient at zero angle of attack, K is factor between the aspect ratio (AR) and Oswald efficiency factor (e) given by K = $\frac{1}{\pi\, e\, AR}$.

The throttle setting express the constraint of maximum power possible. Mathematically is given by:

$$P = \delta_t\, \eta\, P_{max}^s \left(\frac{\rho}{\rho^s}\right)^m ; where\ \delta_t \leq 1.0 \tag{3.25}$$

$$\delta_t = \frac{P}{\eta\, P_{max}^s}\left(\frac{\rho^s}{\rho}\right)^m \leq 1.0 \tag{3.26}$$

While the minimum value of the power for steady level flight can be written as following:

$$P_{min} = \frac{4}{3}\sqrt{\frac{2\,W^3}{\rho\, S}\,\sqrt{3\,K^3\,CD_0}} \tag{3.27}$$

3.3.3 Flight Ceiling

There are two performance parameters, the absolute ceiling and service ceiling. The absolute flight ceiling happens when maximum and minimum power are the same. The service ceiling is the altitude at which the maximum rate of climb is (500 ft/min) or (2.5 m/sec) for jet powered aircraft or (100 ft/min) or (0.5 m/sec) for piston powered aircraft.

3.3.3.1 Jet Aircraft Flight Ceiling

The minimum thrust required for steady level flight, is equal to the maximum thrust produced by the engine at this altitude. This characterizes the flight condition at the flight ceiling. This leads to the mathematical equation:

$$2W\sqrt{K\, CD_0} = T_{max}^s \left(\frac{\rho}{\rho^s}\right)^m \tag{3.28}$$

The numerical solution of this equation gives the air density corresponding to the absolute ceiling. As following:

The previous equation can be re-written as following:

$$error = 2W\sqrt{K\,CD_0} - \; T^s_{max}\left(\frac{\rho}{\rho^s}\right)^m \tag{3.29}$$

To solve this equation, error must become zero. In MATLAB the solution can be found using `fsolve` function, where h is the input, and h is related to ρ value.

3.3.3.2 Propeller Driven Aircraft Flight Ceiling

The minimum powered required for steady level flight, is equal to the maximum power produced by the engine at this altitude. This characterizes the flight condition at the flight ceiling. This leads to the mathematical equation:

$$\frac{4}{3}\sqrt{\frac{2\,W^3}{\rho\,S}\sqrt{3\,K^3\,CD_0}} = \eta\,P^s_{max}\left(\frac{\rho}{\rho^s}\right)^m \tag{3.30}$$

The numeroical solution of this equation gives the air density corresponding to the absolute ceiling.

The previous equation can be re-written as following:

$$error = \frac{4}{3}\sqrt{\frac{2\,W^3}{\rho\,S}\sqrt{3\,K^3\,CD_0}} - \eta\,P^s_{max}\left(\frac{\rho}{\rho^s}\right)^m \tag{3.31}$$

To solve this equation, error must become zero. In MATLAB the solution can be found using `fsolve` function, where h is the input, and h is related to ρ value.

3.3.4 Aircraft Range and Endurance

The aircraft range is the distance that an aircraft covers in steady level flight using a fixed amount of fuel. The aircraft endurance is the time that an aircraft remains in steady level flight using a fixed amount of fuel.

The maximum range and endurance are most important performance figures for each aircraft. The calculation of these two figures differs depending on the type of the engine, jet or propeller. However, the fuel weight and fuel consumption are important factor to the range and endurance.

3.3.4.1 Range of Jet Aircraft

The total distance traveled by Jet aircraft, which is the range R, is obtained by:

$$R = -\int_{W_i}^{W_f} \frac{V\, dW}{c\, T} \tag{3.32}$$

Where, R is the range, W_i is the initial aircraft weight, W_f is the final aircraft weight, T is the thrust produced by the engine, and c is the fuel specific consumption rate.

Using steady level flight assumptions:

- L=W
- T=D
- CL, CD, ρ are constant during the flight.

The range for jet aircraft is given as following:

$$R = \frac{2}{c}\sqrt{\frac{2}{\rho\, S}}\,\frac{CL^{\frac{1}{2}}}{CD}\left[\sqrt{W_i} - \sqrt{W_f}\right] \tag{3.33}$$

Where:

$$c = \frac{TSFC}{3600}\left[\frac{lb_{mass\ of\ fuel}}{sec}\frac{1}{lb_f}\right];\ \text{where TSFC usually is per hour} \tag{3.34}$$

The maximum range R_{max} happens at $\left(\frac{CL^{\frac{1}{2}}}{CD}\right)_{max}$ that can be calculated as following:

$$\left(\frac{CL^{\frac{1}{2}}}{CD}\right)_{max} = \frac{3}{4}\left(\frac{1}{3\, K\, C_{Do}^{3}}\right)^{1/4} \tag{3.35}$$

3.3.4.2 Endurance of Jet Aircraft

The total time that a jet aircraft remains in steady flight, which is the endurance E, is obtained by:

$$E = \int_0^E dt = - \int_{W_i}^{W_f} \frac{dW}{c.T} \tag{3.36}$$

Where, E is the endurance, W_i is the initial aircraft weight, W_f is the final aircraft weight, T is the thrust produced by the engine, and c is the fuel specific consumption rate.

The endurance for jet aircraft can be expressed as following:

$$E = \frac{1}{c}\frac{C_L}{C_D} \ln\left(\frac{W_i}{W_f}\right) \tag{3.37}$$

Where E_{max} happens at $\left(\frac{C_L}{C_D}\right)_{max}$ that can be calculated as following:

$$\left(\frac{C_L}{C_D}\right)_{max} = \frac{1}{2}\sqrt{\frac{1}{K\, C_{Do}}} \tag{3.38}$$

3.3.4.3 Range of Propeller Aircraft

The general expression for the range of a propeller aircraft driven by an ideal internal combustion engine is given by:

$$R = - \int_{W_i}^{W_f} \frac{V\, dW}{cP} \tag{3.39}$$

Where, R is the range, W_i is the initial aircraft weight, W_f is the final aircraft weight, P is the power produced by the engine, and c is the fuel specific consumption rate.

Using steady level flight assumptions:

- L=W
- T=D
- L/D = Cl/Cd = Constant

The range equation can be written as following:

$$R = \frac{\eta}{c}\frac{C_L}{C_D} \ln\frac{W_i}{W_f} \tag{3.40}$$

This equation is known as the *Breguet range* formula for propeller driven aircraft.

Where:

$$c = \frac{TSFC}{550 \times 3600}\left[\frac{lb_{mass\,of\,fuel}}{sec}\frac{1}{lb_f}\right]; \text{ where TSFC usually is in HP and per hour} \tag{3.41}$$

Where R_{max} happens at $\left(\frac{C_L}{C_D}\right)_{max}$ that can be calculated as following:

$$\left(\frac{C_L}{C_D}\right)_{max} = \frac{1}{2}\sqrt{\frac{1}{K\,C_{Do}}} \tag{3.42}$$

3.3.4.4 Endurance of Propeller Aircraft

The total time that a propeller aircraft remains in steady flight, which is the endurance E, is obtained by:

$$E = -\int_{W_i}^{W_f}\frac{dW}{cP} \tag{3.43}$$

Where, E is the endurance, Wi is the initial aircraft weight, Wf is the final aircraft weight, P is the power produced by the engine, and c is the fuel specific consumption rate.

Using steady level flight assumptions:

- L=W
- CL, CD, ρ are constant during the flight.

The endurance formula for propeller driven aircraft is given as:

$$E = -\frac{\eta}{c}\frac{{C_L}^{3/2}}{C_D}\sqrt{2\,\rho\,S}\left[\frac{1}{\sqrt{W_f}} - \frac{1}{\sqrt{W_i}}\right] \tag{3.44}$$

The maximum range E_{max} happens at $\left(\frac{CL^{\frac{1}{2}}}{CD}\right)_{max}$ that can be calculated as following:

$$\left(\frac{CL^{\frac{3}{2}}}{CD}\right)_{max} = \frac{1}{4}\left(\frac{3}{K\,C_{Do}^{1/3}}\right)^{3/4} \tag{3.45}$$

3.3.5 Steady Climbing and Descending Flight

At this phase of the flight, the velocity of aircraft has two components; the vertical and horizontal. The vertical component is called the rate of climb.

The stall constraint for steady flight is expressed as following:

$$\alpha \leq \alpha_{stall}, \tag{3.46}$$

$$V_{stall} \leq V, \tag{3.47}$$

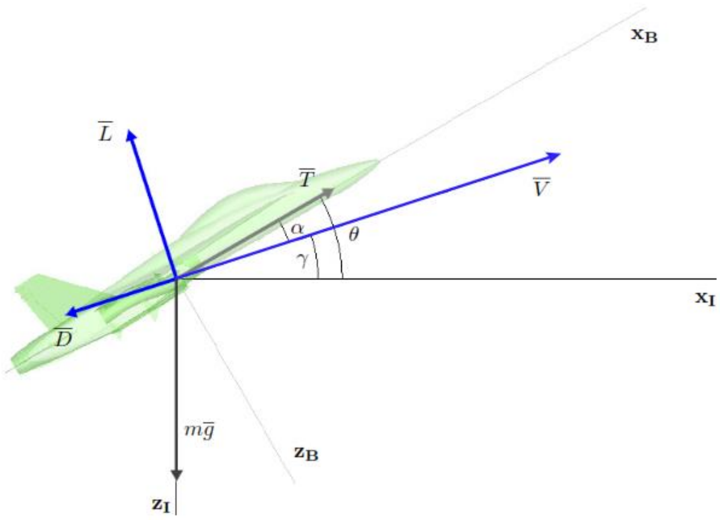

Figure 3.4 Free Body Diagram of an Aircraft in Steady Climbing Flight [24]

The stall angle of attack and stall velocity are given by these equations respectively:

$$V_{stall} = \sqrt{\frac{2\,W}{\rho\,S\,CL_{max}}} \tag{3.48}$$

Where, $C_{L\,max}$ is the maximum lift coefficient, $C_{L\,o}$ is the lift coefficient at zero angle of attack, ρ is the air density at the flight altitude, S is the wing reference area, and W is the aircraft weight.

The second constraint is the thrust/power produced by the engine. A jet engine can provide a maximum rated thrust; and the thrust required for steady level flight must not exceed this maximum thrust. In mathematical tearm, the thrust T required for steady level flight must satisfy:

$$T \leq T^{s}_{max} \left(\frac{\rho}{\rho^{s}}\right)^{m} \tag{3.49}$$

The constrinat on the throttle setting express the constrinat of the ject engine thrust. Mathematically is given by:

$$\delta_t = \frac{T}{T^{s}_{max}} \left(\frac{\rho^{s}}{\rho}\right)^{m} \leq 1.0 \tag{3.50}$$

An internal combustion engine can provide a maximum rated power; and the power required for steady climbing or descending flight must not exceed this maximum power. In mathematical term:

$$P \leq \eta\, P^{s}_{max} \left(\frac{\rho}{\rho^{s}}\right)^{m} \tag{3.51}$$

Where, η is the efficiency of the propeller. The engine power constraint can also be expressed an a constraint on the throttle setting. The engine power constraint is equivalent to the following throttle inequality:

$$\delta_t = \frac{P}{P^{s}_{max}} \left(\frac{\rho^{s}}{\rho}\right)^{m} \leq 1.0 \tag{3.52}$$

3.4 Standard Atmosphere Model

Standard atmosphere model plays a critical role in the study of aircraft performance. In many performance equations mentioned earlier, there are values represents different air characteristics, such as temperature (T), pressure (P), and density (ρ). There are also the ratio of the value at flight level to the value at sea level. Temperature ratio $\theta = T/T_s$, pressure ratio $\delta = P/P_s$, density ratio $\sigma = \rho/\rho_s$. Those atmosphere properties can be obtained from atmosphere tables, or from numerical equations, with small error.

Atmosphere can be naturally divided into several horizontal layers with different physical characteristics. The lowest atmospheric layer is the troposphere where the temperature decreases linearly with altitude. The temperature is 288.15 ^{o}K = 15 ^{o}C at sea level; this layer extends to an altitude of 11 km. In the next layer, referred to as the tropopause, this layer extends from 11 km up to 20 km. The temperature does not change with altitude and stay constant at 216.65 ^{o}K = −56.5 ^{o}C. The next highest layer, the stratosphere extends from 20 km up to an altitude of 32 km. In this region, the temperature increases linearly with altitude. Although it is possible to characterize the temperature variation above the stratosphere, it is not necessary to consider this region for purposes of flight analysis.

Table 3.1 Atmosphere Layers

	SI Units	***US Units***
Sea level	0 km	0 ft
Troposphere	0 km $\leq$ h $\leq$ 11 km	0 ft $\leq$ h $\leq$ 36,089 ft
Tropopause	11 km $\leq$ h $\leq$ 20 km	36,089 ft $\leq$ h $\leq$ 65,616 ft
Stratosphere	20 km $\leq$ h $\leq$ 32 km	65,616 ft $\leq$ h $\leq$ 104,987 ft

The analytical formulas for atmosphere model in each layer are presented below. The derivation of these equations are detailed in many references [24] [25]. The following tables summarize the values and equations of atmosphere model in each layer. The results calculated using these equations are reasonably acceptable.

Table 3.2 Static Values in Atmosphere Model

	SI Units	***US Units***
Sea level	$h_o = 0$	$h_o = 0$
	$T_o = 288.15$ [^{o}K]	$T_o = 518.67$ [^{o}R]
	$P_o = 101{,}325$ [kg/m^2]	$P_o = 2116.2$ [lbs/ft^2]
	$\rho_o = 1.225$ [kg/m^3]	$\rho_o = 2.3769 \times 10^{-3}$ [slug/ft^3]
Troposphere	$h_1 = 11$ [km]	$h_1 = 36{,}089$ [ft]
	$T_1 = 216.65$ [^{o}K]	$T_1 = 389.94$ [^{o}R]
	$P_1 = 22{,}631.9$ [kg/m^2]	$P_1 = 471.85$ [lbs/ft^2]
	$\rho_1 = 0.36392$ [kg/m^3]	$\rho_1 = 0.70494 \times 10^{-3}$ [slug/ft^3]
Tropopause	$h_2 = 20$ [km]	$h_2 = 65{,}616$ [ft]
	$T_2 = 216.65$ [^{o}K]	$T_2 = 389.94$ [^{o}R]

	SI Units	*US Units*
	$P_2 = 5{,}474.9$ [kg/m^2]	$P_2 = 113.95$ [lbs/ft^2]
	$\rho_2 = 0.088035$ [kg/m^3]	$\rho_2 = 0.1702 \times 10^{-3}$ [slug/ft^3]
Constants	$g = 9.80665$ [m^2/sec]	$g = 32.2$ [ft^2/sec]
	$R = 287.0531$ [m^2/sec^2 ^{o}K]	$R = 1716$ [ft^2/sec^2 ^{o}R]
	$a_o = -6.5 \times 10^{-3}$ [^{o}K/m];	$a_o = -3.567 \times 10^{-3}$ [^{o}R/m];
	$a_1 = 1 \times 10^{-3}$ [^{o}K/m]	$a_1 = 0.5494 \times 10^{-3}$ [^{o}R/m]

Table 3.3 Atmosphere Model at Troposphere

SI Units	*US Units*
$\boldsymbol{T = T_o + a_o \times (h \times 10^3)}$	$\boldsymbol{T = T_o + a_o \times h}$, *where h in ft*
$\boldsymbol{P = P_o \times \left(\frac{T}{T_o}\right)^{\frac{-g}{a_o.R}}}$	$\boldsymbol{P = P_o \times \left(\frac{T}{T_o}\right)^{\frac{-g}{a_o.R}}}$
$\boldsymbol{\rho = \rho_o \times \left(\frac{T}{T_o}\right)^{\frac{-g}{a_o.R}-1}}$	$\boldsymbol{\rho = \rho_o \times \left(\frac{T}{T_o}\right)^{\frac{-g}{a_o.R}-1}}$
where h in km	*where h in ft*

Table 3.4 Atmosphere Model at Tropopause

SI Units	*US Units*
$T = T_1 = 216.65\ ^oK$	$T = T_1 = 389.94\ ^oR$
$P = P_1 \times exp\left[\frac{-g}{RT}(h - h_1)\right]$	$P = P_1 \times exp\left[\frac{-g}{RT}(h - h_1)\right]$
$\rho = \rho_1 \times exp\left[\frac{-g}{RT}(h - h_1)\right]$	$\rho = \rho_1 \times exp\left[\frac{-g}{RT}(h - h_1)\right]$
where h in km	*where h in ft*

Table 3.5 Atmosphere Model at Stratosphere

SI Units	***US Units***
$T = T_2 + a_2 \times ((h - h_2) \times 10^3)$ $P = P_2 \left(\frac{T}{T_2}\right)^{\frac{-g}{a_2.R}}$ $\rho = \rho_2 \left(\frac{T}{T_2}\right)^{\frac{-g}{a_2.R}-1}$	$T = T_2 + a_2 \times (h - h_2)$, where h in ft $P = P_2 \left(\frac{T}{T_2}\right)^{\frac{-g}{a_2.R}}$ $\rho = \rho_2 \left(\frac{T}{T_2}\right)^{\frac{-g}{a_2.R}-1}$
where h in km	*where h in ft*

Finally, the speed of sound (a) at each altitude is calculated from following equation:

$$a = \sqrt{\gamma . R . T} \tag{3.53}$$

Where γ = 1.4 is the specific heat of air, R is the universal gas constant, and T is the absolute temperature.

Those values and equations are used in UPA-Gaziantep atmosphere calculation module. In Appendix B, a sample diagrams of results obtained from this module is demonstrated.

CHAPTER 4

METHODOLOGY - SOFTWARE DEVELOPMENT

4.1 Introduction

To achieve research novelty as discussed in the paragraph 2.5, a computer program has been developed with desirable specifications. Those specifications are user friendly, instructive, and open source. In order for the program to be produced with these specifications, various technologies were investigated and then the appropriate ones were selected. The following sections explain each technology adopted.

4.1.1 MATLAB Environment

MATLAB is programing environment widely used in the fields of engineering and applied science including mechanical engineering, electrical engineering, civil engineering, image processing, sound and signal processing, and finance. MATLAB is also widely used for aerospace engineering, and it has dedicated functions and block sets for aeronautics. The name MATLAB is derived from the words (MATrix LABoratory)[26], and as the name indicates, MATLAB is powerful mathematical environment. It performs complicated mathematical operation with simplified commands, making it the perfect option for engineers who wants to solve complicated equations with no deep programming experience. It can manipulate matrices, differentiate and integrate, solve complex space equations by entering few commands. It offers variety of plotting options for vectors, matrices, and even 3D equations. MATLAB has graphical user interface (GUI) that allows MATLAB user to create interactive programs similar to windows applications. These capabilities and other features made MATLAB to be used by millions users in the industry and the academic sectors. As mentioned by (A. Alshehri, 2014) [27] MATLAB was first invented by Cleve Moler in the seventies. In 1984, MATLAB was commercialized by Molder and Jack Little with Steve Bangert, who founded MathWorks [28]. Today, MATLAB is registered trademark for MathWorks.

The MATLAB code filename extension is .m, and because of that, they are called m-files. The m-file is simple text file contains the sequence of commands to be executed. There are two type m-files; the script m-file that is can be executed directly, and the function m-file, that require arguments as inputs to be executed, then run the code within local environment, and return outputs [26].

MATLAB has graphical user interface GUI. It allow the programmer to create windows containing various controls such as menus, inputs, push buttons, and plots boxes. MATLAB GUI is event-driven. That means each control has several events. Each event can trigger execution of the callback function. For example, a push button has `ButtonDownFcn` and `KeyPressFcn`. The GUI environment creates the graphical script automatically and save it with .fig extension. It also creates m-file that contains the callback functions for all controls in the window. The programmer can edit the script in the m-file for each control event [29].

It can be summarized that MATLAB can solve mathematical equations with minimal programing experience, has interactive GUI, and wide spread among students and engineering. These features make MATLAB environment one of the best choices for software developed in this research.

4.1.2 GitHub

GitHub is the largest code sharing and hosting platform. It hosts over 80 million repositories, has 28 million developers, and 1.8 million business and organization as of March 2019 [30]. GitHub inherited its collaborative feature from `git` system. It has the version control system and issue tracking. Kalliamvakou explained the version control system (Kalliamvakou et al., 2014) [31], that it allows users to create their own version from a repository, make changes, and upload it again with pull request to the original code maintainer, who can pull the changes, if appropriate, and integrate it with the main branch of the code. In addition to that, the social features helped GitHub to gain its popularity. Users can create profiles, post their work, comment about their work and receive feedback from other users, who can “follow” or being “followed” others. All of that attract coders, both professional software engineer and amateur ones, to use GitHub for their daily activities.

In their reach about code quality in GitHub projects, Jarczyk et al. [32] describes the work model for Open Source Software. Open Source Software is often developed by volunteering virtual teams, for that they cannot adapt the traditional management. Instead, the code development processes is managed via collaborative networks such as GitHub. Those networks compensate for the traditional centralized management by “version control” system and social features. Users can join/leave teams, initiate or develop specific parts of the code, and all the work is being consolidated into one functioning code.

Since this work is intended to create software accessible by students and engineers, and possible to be developed, than the resulting software must be free of charge, and open source for collaboration and development. From what have been reviewed above, it becomes obvious that the best choice for the software developed in this research is to be a free open source software (FOSS) hosted online at collaborative site such as GitHub.

4.2 Description of UPA-Gaziantep

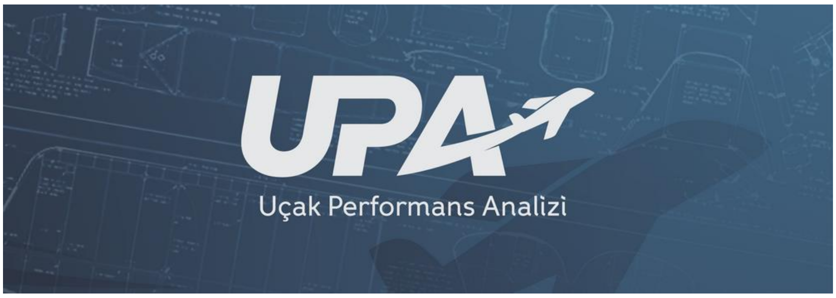

Figure 4.1 UPA Logo

UPA-Gaziantep is an open-source user-friendly computer program for the aircraft performance, developed in Gaziantep University, created in MATLAB 2009 and MATLAB GUI. It can be used as educational software for aeronautical students to be used aircraft performance curricula such as Flight Mechanics and Aircraft Stability and Control. The program name has been derived from the initials of (Uçak Performans Analizi) which means (Aircraft Performance Analysis) in Turkish Language. To avoid any confusing with any other meaning of the acronym UPA, the word Gaziantep has been added to the software.

UPA-Gaziantep uses the physical configurations of the aircraft and power plant specification as an input to analyze and plot main performance parameters, depending on mathematical models and experimental databases. In the current version, UPA-Gaziantep has modules that calculate the following point-performance cases: Steady Level Flight, Steady Climb Flight, Gliding Flight, Range, and Endurance.

4.3 Programming Approach

4.3.1 Program Flowchart

The program consists of the following parts. 1) Input and input-processing components. 2) Mathematical calculations components. 3) Results displaying components.

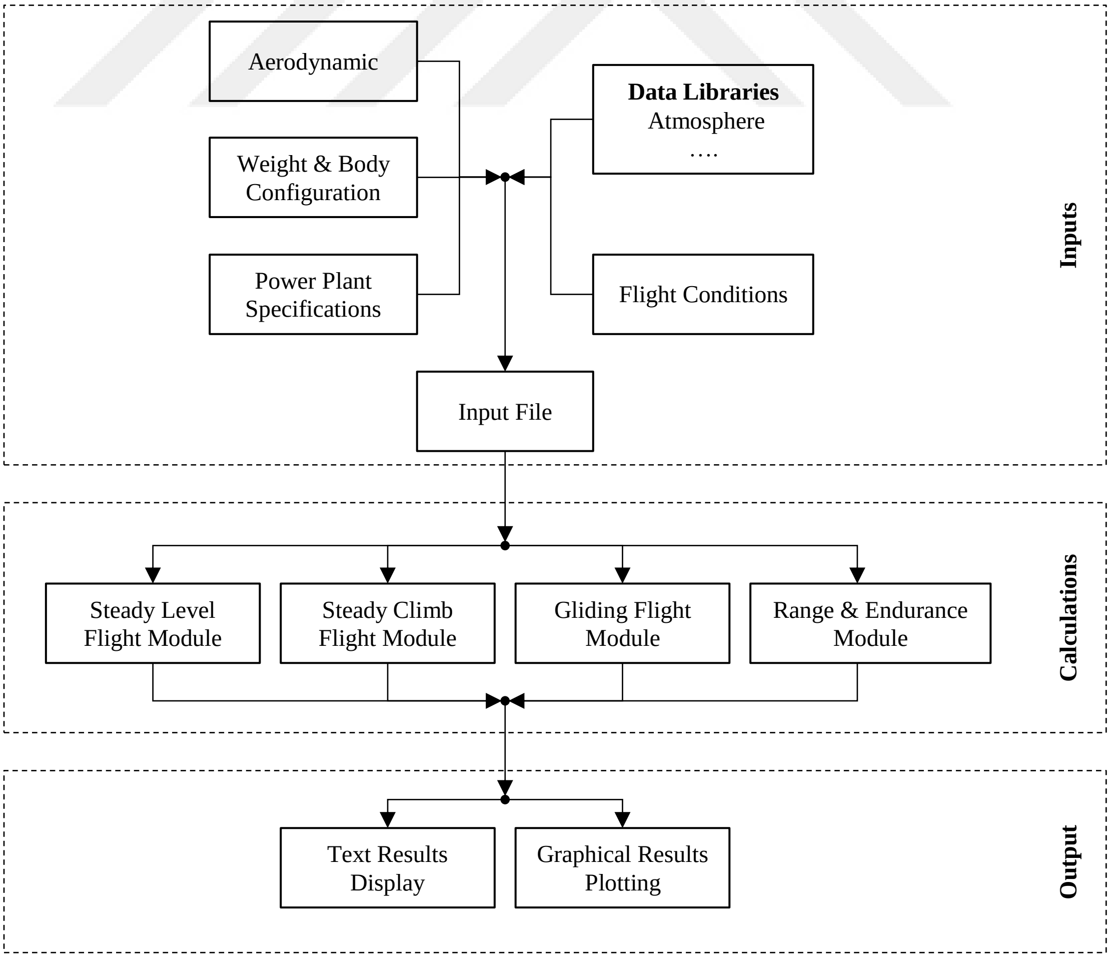

Figure 4.2 UPA-Gaziantep Flowchart

The inputs are required to be entered by the program user. Some inputs are direct values like the aircraft weight. While some inputs are indirect values, that help calculating the direct values from databases or from mathematical equations. Atmosphere parameters are one example that are being calculated from the aircraft altitude. After getting all inputs values ready, the software combine all inputs values into Input File. The user can choose from the software options the case required to be calculated. Based on user choice, the inputs file is used to calculate one or more of the aircraft performance parameters. Then, the results can be displayed as text or in plot forms.

4.3.2 Program Modules

As have been displayed in the program flowchart, the program consists of calculation modules. Each module solve part of the performance problem. The modular architecture make the code more readable and possible to edit/develop.

The equation in each module has been discussed in chapter 3. Main modules for performance calculations are atmosphere module, cruise flight phase, glide flight phase, climb and decent flight phases, range and endurance. In addition to the performance calculations, there are functional modules such as display module, and unit dictionary.

4.3.3 Naming Conventions

All names used in the program must be descriptive to make it easy to read by program user and indicate the meaning or function of the named item. This applies to variables, files, folders, and functions. For example, the variable CL_alpha refers to C_{l_α}, and MTOW refers to the aircraft Maximum Take-off Weight. In the same way, the file `eqnOswaldFactor.m` includes the equations for Oswald Factor calculation.

4.3.4 Code Documentation

To enable users to read and understand the code, the code has been documented as much as possible. Comments are placed in two positions, the file header, and inline.

The header comments contain the information about the file and its inputs and outputs. The inline comments explain the function of the same line. The comments explains the function of each part of the code. Also for essential equations, the inline comments contain the equation number from the theoretical study in this thesis or from reference.

4.3.5 Directory Structure

The program files have been organized within folder similar to the structure of open source software. The directory structure is kept as simple as possible to allow further development. below is preview of the directory structure.

```
├── ...
│
├── /docs                         # Documentation files
│
├── /src                          # Program Source files
│   ├── /eqns                     # Performance Equations
│   ├── /GUIsubroutines           # GUI Subroutines
│   ├── /MATLAB_Functions         # Non-Aeronautical Functions
│   ├── /lib                      # Program Libraries
│   ├── /StdAtm                   # Standard Atmosphere Module
│   │
│   ├── FlightPhase.m             # Performance Equations of FlightPhase
│   └── ...
│
├── /input                        # Input files
│   └── input_AircraftName.m      # Input file for AircraftName
│
├── main.m                        # Script of main routine
├── ...
├── LICENSE.txt                   # GNU General Public License
└── README.txt                    # Instruction to run the program
```

Figure 4.3 Directory Structure

4.4 Running the Program

4.4.1 Installation

The latest copy of the source code is always updated at the UPA-Gaziantep Repository in GitHub. It can be downloaded the link https://github.com/UPAGaziatep/UPAGaziantep. The downloaded zip package must be extracted and copies to MATLAB "current folder". Alternatively, it can be extract the package in a different location, the MATLAB "current must" be changed to UPA-Gaziantep path.

UPA-Gaziantep requires MATLAB R2017a or newer versions. The code was not tested on older versions where compatibility issues might arise. MATLAB are available on Windows, iOS, and Linux version.

The program can be run by entering the command (`main`) to Command Window. The program displays the main menu as shown in Figure 4.4.

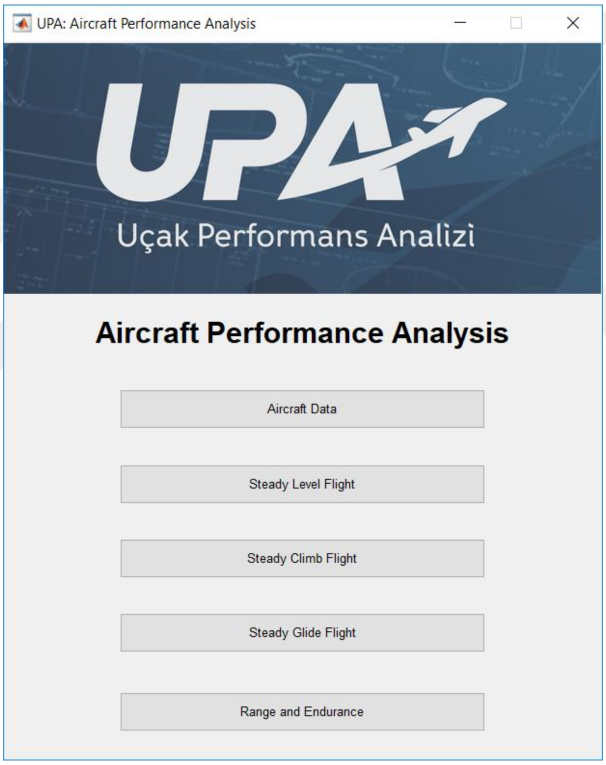

Figure 4.4 UPA-Gaziantep Main Menu

4.4.2 Data Input

The first button on the main menu is the (Aircraft Data) menu for entering aircraft information. The input are structured into the following groups: units, weight, configuration and structure, power plant, and aerodynamics. Each group contain set of data required. There is also optional inputs such as Oswald factor. The user can leave them blank, and the program can estimate them form equation or data library.

The program accepts both ‘SI’ and ‘US’ units. The program automatically select the appropriate equations and show the related unit for results.

4.4.3 Getting Results

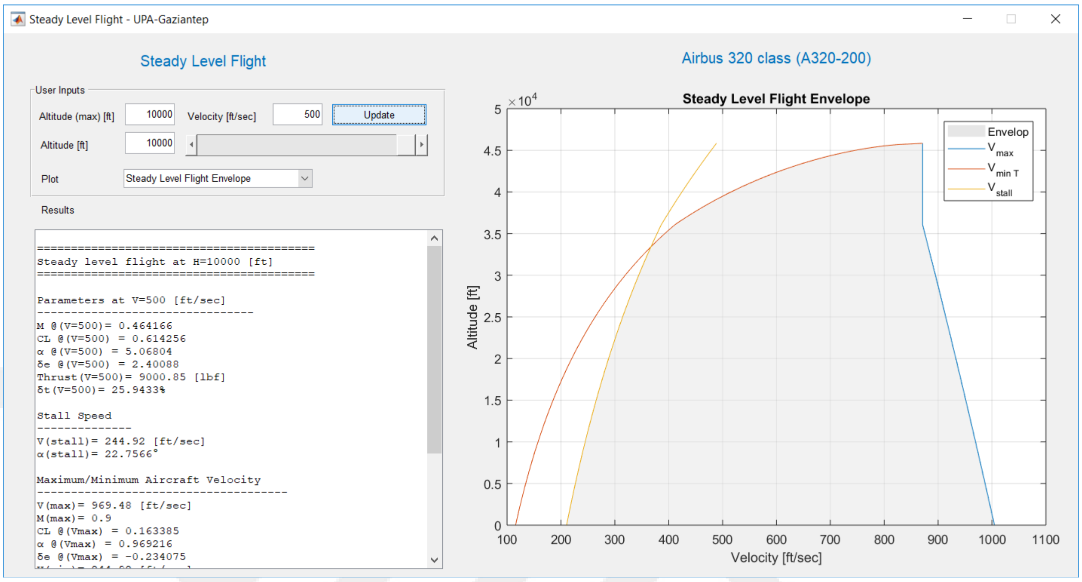

Figure 4.5 Steady Level Flight Results Window

The main menu shows the several phase analysis options. Each button opens the results window for each phase. In results window, the user has control for phase inputs, such as flight altitude for cruise flight, or path angle γ for climb steady flight.

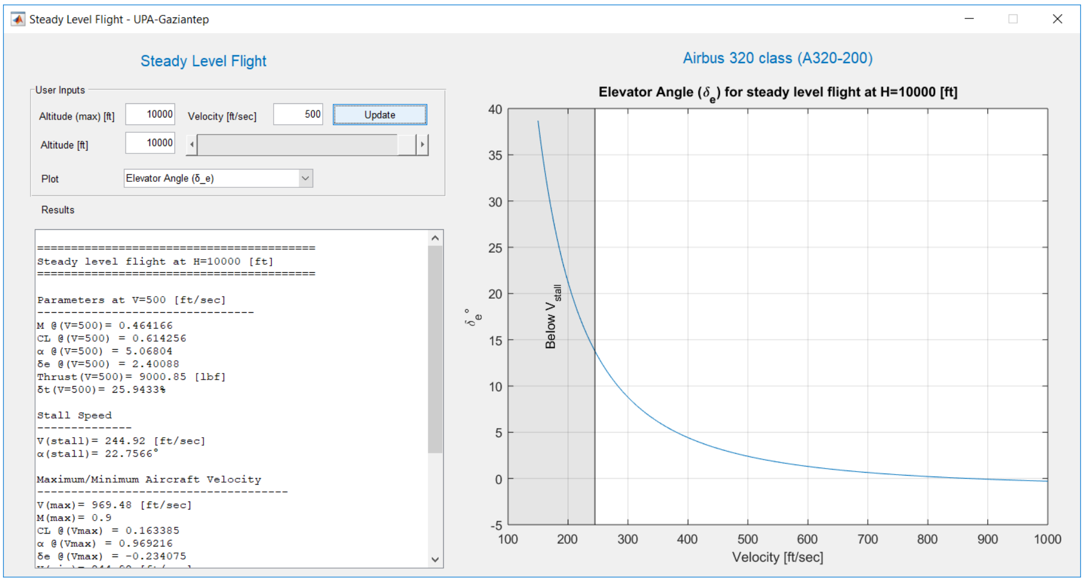

Figure 4.6 Elevator Angle δ_e vs. Velocity Plot

The results are displayed with two formats on same window as shown on Figure 4.5; text results and plotted results. The text results contains all main parameters for the phase being analyzed. The (Plot) drop list change the type of diagram to be displayed. For example, for Steady Level Flight, the program can display flight envelop, trust required vs Velocity, Angle of Attach (AoA) α vs. Velocity, and Elevator Angle δ_e vs. Velocity.

To validate the results of the program; a case study is presented in the following chapter. The results from the case study are discussed to reach thesis conclusion.

CHAPTER 5

VALIDATION CASE-STUDY

5.1 Introduction

This chapter explore the validity of UPA-Gaziantep, by running two case studies on the program. One for jet aircraft and one for propeller aircraft. Despite the fact that the program is designed for aircraft during design process, but for the research purpose, the case study chooses an existing aircraft where all aircraft specification and performance parameters are determined and available for comparison.

In the first case study, Airbus 320 Class (A320-200) is used. In the second, Cessna 172 Skyhawk is used. The inputs required for the program are looked-up and made ready. Those inputs are entered into the program to get results. The results from the program are compared with aircraft well-known specification.

5.2 Case Study Aircraft: Airbus A320-200

The famous jetliner, Airbus A320-200 is a member of A320 family. The A320 was the first narrow body airliner to use an appreciable amount of composite materials in its construction [33] and the first civil aircraft to pioneer fly-by-wire technology. It has set the standard ever since its arrival and, thanks to significant annual investment of 300 million euros, the A320 Family continues to innovate [34]. It has With capacity of 150-159 passengers. Turkish Airlines feet has 22 Airbus A320-200 according to the company website [35]. The A320 family configured as a low wing monoplane with a cantilevered wing with a sweep back of 25 degrees. The single-aisle narrow-body jet is powered by two engines located one under each wing [33].

Figure 5.1 Airbus A320-200 Turkish Airlines (THY) – BEYKOZ [36]

5.2.1 Airbus A320-200 Inputs

The software inputs are the aircraft specifications that can be described in the parameters of weight, geometry, power plant, and aerodynamics. Since the aircraft A320-200 exists; its parameters are published on different sources including official website of AIRBUS, aircraft design books and related papers. In the following paragraphs, the A320-200 parameters are collected and arranged to be used as software inputs.

5.2.1.1 Aircraft Weight

The weight information for flight mission are found Appendix D of the book Aircraft Design by Kundu [37]. The fuel consumption mentioned in Table 5.2 is calculated for maximum flight possible by the aircraft, to be able to calculate maximum range and endurance possible by the aircraft.

Table 5.1 A320-200 Weight Information [37]

Specification	***in US Units***
Maximum Take-off weight (MTOM)	162,000 lb
Onboard fuel mass (W_{fuel})	40,900 lb
Payload (200 lb × 150 pax)	30,000 lb

Table 5.2 A320-200 Fuel Consumption for Flight Mission Sector [37]

Sector	***Fuel consumed (lb)***
Taxi out	200
Takeoff	300
Climb	4,355
Cruise (Maximum)	28,400
Descent	370
Approach/land	380
Taxi in	135
Total	**34,140**

From the detail in those tables, we can extract the following values:

Maximum Take-off weight (MTOM) = 162,000 lb

Weight of fuel for flight mission (W_f)= 34,140 lb

Initial cruise weight ($W_{cruise\ i}$) = MTOM – taxi-out – take-off – climb

Initial cruise weight ($W_{cruise\ i}$) = 162,000 – 200 – 300 -4,355 = 157,145 lb

Finial cruise weight ($W_{cruise\ f}$) = $W_{cruise\ i}$ – Cruise fuel

Finial cruise weight ($W_{cruise\ f}$) = 157,145 – 28,400 = 128,745 lb

The above calculation results in W_i and W_f required for Maximum Range calculations, as following:

For Maximum Range: W_i = 157,145 [lb], and W_f = 128,745 [lb].

5.2.1.2 Aircraft Geometry

The geometry specification are found on the same reference mentioned in the section above.

Table 5.3 A320-200 Geometry Information [37]

Specification	***in US Units***
Length	123 ft 3 in
Height	38 ft 7 in
Wing Span (*b*)	111 ft 10 in
Wing Area	1320 ft^2
Wing Reference Area (*S*)	1,202.5 ft^2
Aspect Ratio (*AR*)	9.37
Limit Load Factor (n_{max})	2.6

5.2.1.3 Power Plant

Figure 5.2 CFM56-5B engine, Used in Airbus A320 Family [38]

AIRBUS brochure (A319/A320/A321 Flight deck and systems briefing for pilots) [39] mentions that A320 aircrafts have two options for power plant. The CFM International turbofan engines (CFM56-5A1), and International Aero Engines IAE (V2500-A1). In this case study, the data for CFM56-5A1 is used.

Table 5.4 A320-200 Engine Information [39]

Specification	***in US Units***
Propulsion	2 turbofan engines
Engine model	CFM56-5A1
Engine power (each)	25,000 lbf
Installed sea-level static thrust (each)	23,500 lbf
Trust Specific Fuel Consumption (TSFC)	0.5648 (lbs/hr)/lbf

For this case study, the thrust required in the thrust for two installed engines. That means: Thrust = 23,500 × 2 = 47,000 [lbf]

5.2.1.4 Aerodynamics

Usually for commercial aircrafts, manufacturing companies do not state the airfoil used for wing and tail. Most likely because of competition for best performance. Anyway, the A320 airfoil was mentioned in paper published by Karukana [40] the airfoil of A320 is identified as NACA 23015.

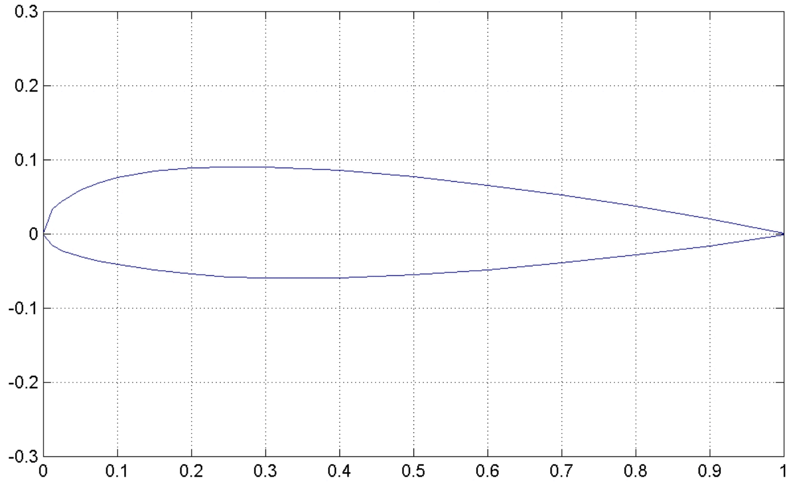

Figure 5.3 NACA 23015 [41]

From aerodynamic data libraries in Civil Jet Aircraft Design [4] and the data for aircraft speed is being found in aircraft data file [42], and from Aircraft Design by Kundu [37], the aerodynamics are determined.

Table 5.5 A320-200 Aerodynamics Information

Specification	**Value**
$C_{L_{max}}$	2.56
C_{L_o}	0.05677
C_{L_α}	0.11 [degree^{-1}]
C_{D_o}	0.0213
k In polar equation ($C_D = C_{D_o} + k.C_L^2$)	0.034

5.2.2 Airbus A320-200 Performance Parameters

The following critical parameters are used in this case study. Flight speed, range, and service ceiling.

The data for aircraft speed is being found in aircraft data file [42] from Civil Jet Aircraft Design [4]. The details for range and service ceiling are found on commercially information available about A320 [43].

Table 5.6 A320-200 Performance Information

Specification		
V_2	145 [knots]	245 [ft/sec]
V_{app}	134 [knots]	226 [ft/sec]
V_{mo}/M_{mo}	350 [knots] / M0.82	590 [ft/sec] / M0.82
V_{ne}/M_{ne}	381 [knots] / M0.89	643 [ft/sec] / M0.89
Service Ceiling	39,800 ft	
Range	3,542 mile	

The stall velocity relates to the V_2 with the following equation:

$$V_2 = 1.2 * V_{stall} \quad at\ (h = 0) \tag{5.1}$$

Based on this equation, the design V_{stall} can be calculated as following:

$$V_{stall}(@\ h = 0) = \frac{V_2}{1.2} = \frac{245}{1.2} = 204.17\ [ft.sec^{-1}\ ] \tag{5.2}$$

The data (inputs and performance parameters) for this case study is used in the following chapter for validation and discussion.

5.3 Case Study Aircraft: Cessna 172 Skyhawk

Figure 5.4 Cessna Skyhawk [44]

Cessna 172, which is also known as Skyhawk, is a four-seat, single-engine, high wing, fixed-wing aircraft. It is made by Cessna Aircraft Company, which became brand of by Textron Aviation since 2014. The first Cessna 172 was first produced in 1956, and it is still in production today. More than 43,000 Cessna 172s have been produced, for that it can be considered as the one of the most successful light aircraft series [45]. During the years, Cessna 172 has many modifications such as; engine with additional power, modified propeller, additional fuel tanks in wing tips, and wheel fairing to reduce drag. These remarkable features make this aircraft a great case study in this research.

5.3.1 Cessna 172 Inputs

Due the multiple variants of Cessna 172, there are different specification of the aircraft. In this case study, the specification of Cessna 172 Skyhawk II/100 are used, as per mentioned in the Performance Assessment by Mciver, 2003 [46].

5.3.1.1 Aircraft Weight

The Cessna 172 specification are quoted from Jane's All the World's Aircraft 1977-98 [46] as the following:

Table 5.7 Cessna 172 Weight Information [46]

Specification	*in US Units*
Maximum Takeoff Weight	2300 lbs.
Weight – Empty Equipped	1403 lbs.
Baggage Capacity	120 lbs.
Fuel Capacity	43 US gallons 258.86 lbs. *
Useable Fuel Capacity	38 US gallons 228.76 lbs. *
Maximum Wing Loading	13.2 lbs./sq ft.
Maximum Power Loading	14.4 lbs./hp

* Calculated as the weight of 100LL avgas is 6.02 lbs per US gallon

Fuel consumption weight varies with flight altitude. Table 5.8 and Table 5.9 show the weight of fuel consumed in each flight segment for flight at 8,000[ft] and 10,000 [ft] respectively.

Table 5.8 Fuel consumption for flight altitude 8,000 [ft] for Cessna 172

Specification	*in US Units*
Taxi/Take-off fuel used	5.88 lbs
Climb fuel used	19.51 lbs
Descent fuel used	24.06 lbs
Landing/Taxi fuel used	4.20 lbs
Reserve fuel available	56.70 lbs
Total fuel for climb and descent	110.36 lbs
Fuel available for cruise	112.70 lbs

Table 5.9 Fuel consumption for flight altitude 10,000 [ft] for Cessna 172

Specification	***in US Units***
Taxi/Take-off fuel used	5.88 lbs
Climb fuel used	24.39 lbs
Descent fuel used	30.08 lbs
Landing/Taxi fuel used	4.20 lbs
Reserve fuel available	56.70 lbs
Total fuel for climb and descent	121.25 lbs
Fuel available for cruise	101.81 lbs

From the information in the tables above, the initial and final weight can be calaculated for each flight altitude as following:

Maximum take-off weight (MTOW) = 2,300 lb

Fuel weight for complete flight mission = 121.25+101.81 = 223.06 lb

Initial cruise weight ($W_{cruise\ i}$) = MTOW – taxi – climb

Finial cruise weight ($W_{cruise\ f}$) = $W_{cruise\ i}$ – Cruise fuel

$W_{cruise\ i}$ (FL=8,000 ft) = 2,300 – 5.88 – 19.51 = 2,274.61 lb

$W_{cruise\ f}$ (FL=8,000 ft) = 2,274.61 – 112.70 = 2,161.91 lb

$W_{cruise\ i}$ (FL=10,000 ft) = 2,300 – 5.88 – 24.39 = 2,269.73 lb

$W_{cruise\ f}$ (FL=10,000 ft) = 2,274.61 – 101.81 = 2,167.92 lb

5.3.1.2 Aircraft Geometry

From same reference mentioned in weight section, the geometry specification are the following:

Table 5.10 Cessna 172 Geometry Information [46]

Specification	***in US Units***
Length overall	26 ft 11 in
Height overall	8 ft 9.5 in
Wing area	174 sq ft (gross) 173 sq ft (input)
Wing span	35 ft 10 in
Wing root chord	5 ft 4 in
Wing tip chord	3 ft 8.5 in
Wing aspect ratio	7.52
Tail plane span	11 ft 4 in
Aileron Area	18.3 sq ft
Flap Area	21.2 sq ft
Vertical Tail Area	11.24 sq ft
Rudder Area	7.43 sq ft
Horizontal Tail Area	21.56 sq ft
Elevator Area	14.53 sq ft

5.3.1.3 Aircraft Engine

Figure 5.5 A Lycoming O-320-D2A installed in a Symphony SA-160 [47]

To secure flight power, the Cessna 172 uses single piston engine with fixed-pitch propeller. Different engines has been used with this aircraft along the time. For that, different specifications can be found when searching for Cessna 172 engine type and power. In earlier models, the engine has power of 160 hp. While in the modern ones, Cessna website [44] state the engine power is 180 hp. The specification used in this case study are shown in Table 5.11, while the photo above shows similar engine.

Table 5.11 Cessna 172 Engine Information

Specification	***in US Units***
Engine Power	160 hp
Engine Type	Lycoming O-320-H max power at 2700 rpm
Propeller	Two bladed fixed pitch metal propeller
Propeller diameter	6 ft 3 in
Average Range Propulsive Efficiency	0.70

5.3.1.4 Aircraft Aerodynamics

Regarding the aerodynamics of The Cessna 172, it uses NACA 2412 airfoil for the wings, NACA 0012/0009 airfoils for horizontal tail root/tip, and NACA 0009/0006 for airfoils for vertical tail root/tip

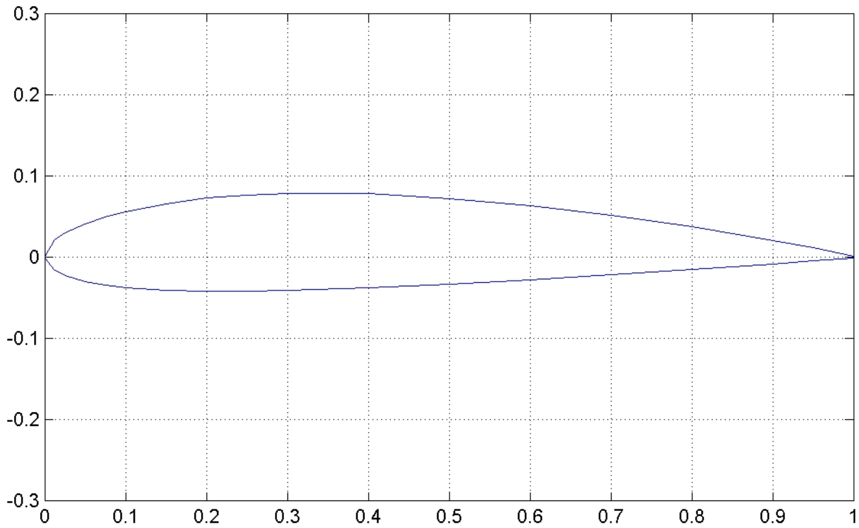

Figure 5.6 NACA 2412 [41]

The analysis conducted by Mciver, 2003 [46] shows the approximation for Cessna 172 aerodynamic parameters, based on several references including Fluid Dynamic Drag by Hoerner [48]. Lift coefficients are estimated for NACA 2412.

Table 5.12 A320-200 Aerodynamics Information

Specification	**Value**
$C_{l_{max}}$ (no flaps)	1.60
$C_{l_{max}}$ (with flaps)	2.10
C_{L_o}	0.02
C_{l_α}	0.12 [degree^{-1}]
C_{D_o}	0.0341
k In polar equation ($C_D = C_{D_o} + k.C_L^2$)	0.0554

5.3.2 Cessna 172 Performance Parameters

The following critical parameters are used in this case study. Flight speed, range and Endurance. The performance parameters are mentioned in Mciver, 2003 [46].

Table 5.13 Cessna 172 Performance Information

Specification		
Never Exceed Speed	174 mph	255.2 ft/sec
Max Level Speed (at SL)	144 mph	211.2 ft/sec
Max Cruising Speed (75% power at 8000 feet)	140 mph	205.4 ft/sec
Stalling Speed (flaps up)	57 mph	83.6 ft/sec
Stalling Speed (flaps down)	51 mph	74.8 ft/sec
Service Ceiling	14,200 feet	
Range at 8,000 feet cruise	584 miles	
Range at 10,000 feet curies	539 miles	
Endurance at 8,000 feet cruise	3 hr : 48 min	
Endurance at 10,000 feet cruise	3 hr : 36 min	

The data for case study prepared in this chapter, enable the validation of UPA-Gaziantep, by using the input information, and compare the results with performance parameters. This is covered in the following chapter.

CHAPTER 6

VALIDATION RESULTS AND DISCUSSION

6.1 Methodology

The validation processes is run via imitation of aircraft design processes, where aircraft design inputs are determined; so the software can provide the performance of this aircraft. Since the aircraft already exists, so the results should match the aircraft performance published from the manufacturer.

The data of the case study (inputs/outputs) are prepared in the previous chapter. In this chapter, the result are obtained and then discussed.

6.2 Results Error

To compare the results obtained from the program with the aircraft performance parameters published from the manufacturer, the error is to be calculated.

According to Taylor [49], the percentage error for resulted values is calculated from the following equation:

$$percentage\ error = \left|\frac{calculated\ vale - reference\ value}{reference\ value}\right| * 100\ (\%) \tag{6.1}$$

6.3 Analyzing Airbus A320-200

The inputs for A320-200 are entered into UPA-Gaziantep. For each performance parameters, the relevant velocity and altitude data is being set as inputs, and results are obtained.

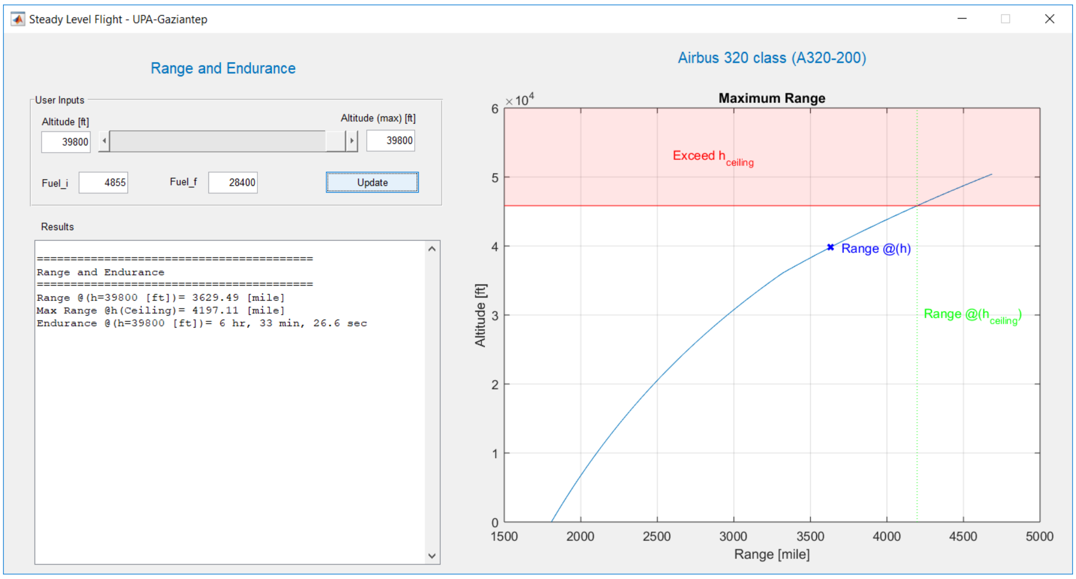

Figure 6.1 UPA-Gaziantep Results Window

6.3.1 Stall Velocity

As has been mentioned before, the stall velocity relates to the V_2 with the following equation: $V_2 = 1.2 * V_{stall} \quad at\ (h = 0)$

The program can calculate V_{stall} from (Steady Level Flight) results at (h=0). The obtained results shows that V_{stall} = 210.42 [ft/sec]

```
=========================================
Steady level flight at H=0 [ft]
=========================================
Stall Speed
--------------
V(stall)= 210.428 [ft/sec]
α(stall)= 22.7566°
```

The reference value for stall velocity V $_{stall}$ (@h=0) is 204.17 [ft/sec], and the calculated value is 210,4 [ft/se]. The error can be calculated for the following:

$$\%\ error_{(V_{stall})} = \left|\frac{210.43 - 204.17}{204.17}\right| * 100 = 3.07\% \tag{6.2}$$

This results means that, the results obtained from UPA-Gaziantep has 3.07% error.

6.3.2 Cruise Range

Aircraft curies range is calculated at Service Ceiling (39,800 ft). The (Range and Endurance) results obtained from the program at altitude of (h=39,800 ft) shows that Cruise Range = 3629.5 [mile]

```
=========================================
Range and Endurance
=========================================
Range @(h=39800 [ft])= 3629.49 [mile]
Endurance @(h=39800 [ft])= 6 hr, 33 min, 26.6 sec
```

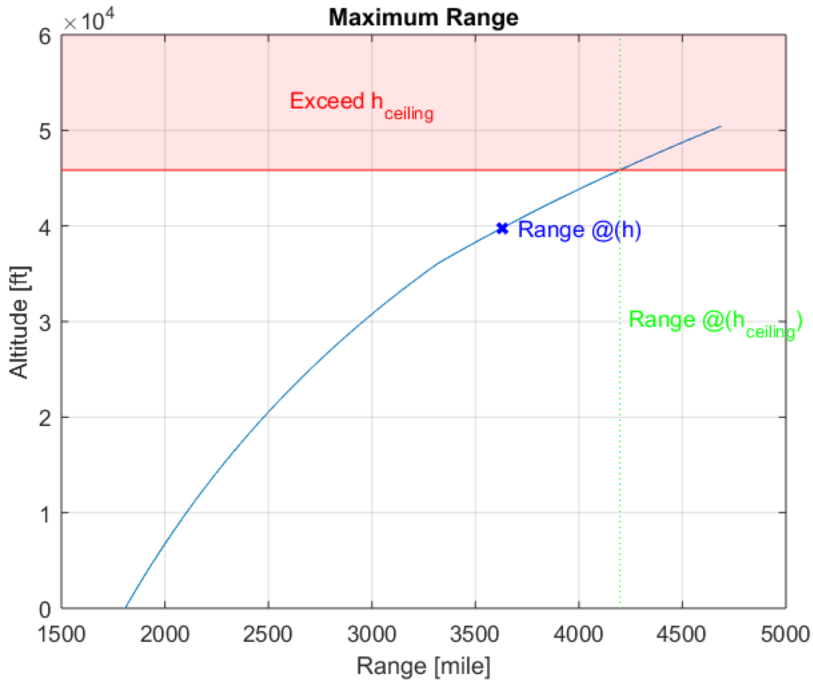

Figure 6.2 Range Diagram from UPA-Gaziantep

The reference value for cruise range is 3,542 [mile], and the calculated value is 3,629.5 [mile]. The error can be calculated for the following:

$$\% \, error_{(Range)} = \left| \frac{3{,}629.5 - 3{,}542}{3{,}542} \right| * 100 = 2.47\% \tag{6.3}$$

This results means that, the results obtained from UPA-Gaziantep has 2.47% error.

6.3.3 Service Ceiling

The service ceiling is the altitude at which the maximum rate of climb is 500 [ft/min]

(2.5 m/s) for jet powered aircraft, or 100 [ft/min] (0.5 m/s) for piston powered aircraft.

The program calculates the Rate of Climb at each point from 0 to maximum altitude (Max ceiling) and finds the service ceiling point matches the later mentioned rate of climb. The following results are obtained from the program.

```
=========================================
Max Ceiling and Service Ceiling
=========================================
Max Ceiling = 45839.1 [ft]
Service Ceiling = 42265.4 [ft]
```

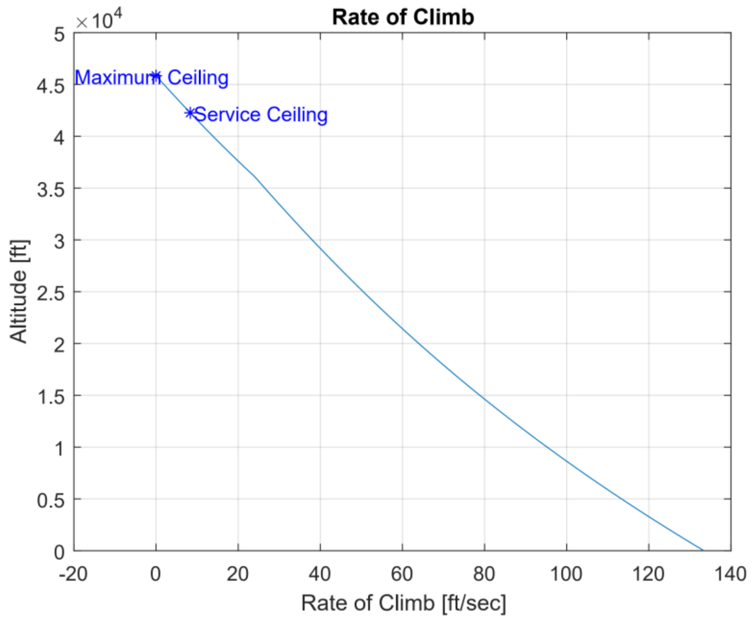

Figure 6.3 Max Ceiling and Service Ceiling from UPA-Gaziantep

The reference value for Service Ceiling is 39,800 [ft], and the calculated value is 42,265.4 [ft]. The error can be calculated for the following:

$$\%\ error_{(Service\ Ceiling)} = \left|\frac{42{,}265.4 - 39{,}800}{39{,}800}\right| * 100 = 6.19\% \qquad (6.4)$$

This results means that, the results obtained from UPA-Gaziantep has 6.19% error.

6.4 Analyzing Cessna 172

The inputs for Cessna 172 Skyhawk are entered into UPA-Gaziantep. For each performance parameters, the relevant velocity and altitude data is being set as inputs, and results are obtained.

6.4.1 Stall Velocity

The program can calculate V_{stall} from (Steady Level Flight) results at (h=0). There are two cases, one with flaps up (C_L=1.6), one with flaps down (C_L=2.1). Below is the results for each

```
==========================================
Steady level flight at H=0 [ft]
==========================================
Stall Speed
--------------
V(stall)= 83.6162 [ft/sec]
M(stall)= 0.074907
CL @(Vstall) = 1.6
α @(Vstall) = 13.1667°
δe @(Vstall) = 12.4444°
```

```
==========================================
Steady level flight at H=0 [ft]
==========================================
Stall Speed
--------------
V(stall)= 72.9862 [ft/sec]
M(stall)= 0.0653842
CL @(Vstall) = 2.1
α @(Vstall) = 17.3333°
δe @(Vstall) = 16.8889°
```

Results show that V_{stall} (flaps up) = 83.62 [ft/sec], V_{stall} (flaps down) = 72.99 [ft/sec]

The reference value for stall velocities are:

Stalling Speed (flaps up)	57 mph	83.6 ft/sec
Stalling Speed (flaps down)	51 mph	74.8 ft/sec

The error for each case can be found as the following

$$\%\ error_{(V_{stall})}(falps\ up) = \left|\frac{83.62 - 83.6}{83.6}\right| * 100 = 0.02\% \tag{6.5}$$

$$\% \; error_{(V_{stall})}(falps \; down) = \left|\frac{72.99 - 74.8}{74.8}\right| * 100 = 2.42\% \quad (6.6)$$

The results above can indicate that results obtained from UPA-Gaziantep has a maximum error of 2.42%.

6.5 Results Discussion

The results obtained from UPA-Gaziantep program, that are calculated in the previous paragraphs, indicates that an error less than 7%. Based on the maximum error, the program accuracy can be calculated.

Maximum error is 6.19%. Thus, the accuracy is 100-6.19 = 93.81%

UPA-Gaziantep Accuracy = 93.81%

Taking in consideration that; 1) Inputs can vary from real aircraft parameters due to lack of accurate source of information such as wind-tunnel inputs, 2) UPA-Gaziantep requires minimum inputs leading to approximation in calculation and 3) UPA-Gaziantep adopts simplified mathematical models, It can be easily stated that: UPA-Gaziantep results offers an acceptable performance estimation during preliminary aircraft design stage for educational and research purposes.

CHAPTER 7

CONCLUSION

The aircraft performance calculation is complicated and time-consuming process. For that; many computer solutions have been developed. Most available software are costly commercial software; thus not accessible by students, or software was written in FORTRAN and cannot be developed easily. UPA is an open-source MATLAB program developed in Gaziantep University that calculate the aircraft performance parameters. The results of UPA-Gaziantep, the program developed in this research, are validated via case studies of the commercial jet aircraft (Airbus A320-200) and Cessna 172. The results showed that the program delivers results with an accuracy of 93%. This accuracy is within the acceptable range of approximation due to the limited number of inputs, and simplification of mathematical model. This indicates that UPA-Gaziantep results can be useful for preliminary decision only. The students and aircraft-designers in Turkey and outside can use this software for designing and learning objectives. Nevertheless, the software requires further development and further validation to include other types and show more results.

FUTURE WORK

This research is meant to initiate a crowd-source aircraft performance software, that can be used and updated by many students and enthusiastic engineers and hobbyist around the globe. The work done represents only one step in a long journey. While many steps are still required to reach that dream.

The software is still in need for a lot of development. The following areas of work are only suggestions, but one may suggest others as well;

- Conducting more validation to the results.
- Adding more flight phases to the calculations.
- Adding more aircraft types to the program.
- Adding more libraries

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APPENDIX A: UPA-GAZIANTEP CODE

Filename: code_main.m

```
% UPA-G - Uçak Performans Analizi Gaziantep
% Aircraft Performance Analysis Program
%==========================================
% Last updated: 2018-09-07

%====================================
%       PROJECT INITILIZATION
%====================================

restoredefaultpath;
addpath(genpath('.'));
clc;clear;close all;clear all;
format

%====================================
%       AIRCRAFT INITILIZATION
%====================================
% Read Aircraft Inputs
Aircraft_Inputs = Read_A320_US();
% Create New Aircraft Class
Aircraft = AircraftClass(Aircraft_Inputs);
% Read all Text from dictionary
lib_Dictionary

%====================================
%        Main Calculations
%====================================
% Flight State Inputs
% (h),(V),(gamma) are flight state inputs
FlightStateInputs.h = 39800;        % altitude [m][ft]
FlightStateInputs.V = 200;          % Velocity [m/sec][ft/sec]
FlightStateInputs.gamma = 10;       % flight path angle [deg]

%
%====================================
% Ceilings
%====================================
% Calculations
ceilings = Ceiling(Aircraft);
% Display
disp(ceilings.ResultsText)
% Plotting
ceilings.Plot_RC;
%}

%
%==========================================
% Steady Gliding Flight at (h,V,gamma)
%==========================================
% Calculations
GlidingCase = SteadyGlidingState(Aircraft,FlightStateInputs);
% Display
disp(GlidingCase.ResultsText())
% Plotting
SteadyGlidingPlot(Aircraft,FlightStateInputs,0)
%}

%
%====================================
% Steady Level Flight at (h,V)
%====================================
% Calculations
LevelCase = SteadyLevelState(Aircraft,FlightStateInputs);
% Display
disp(LevelCase.ResultsText())
% Plotting
SteadyLevelPlot(Aircraft,FlightStateInputs,0)
%}

%
%====================================
```

```
% Aircraft Range and Edurance (h)
%====================================
FlightStateInputs.W_fuel_i = 200+300+4355;  % Weight of fuel till start of cruise
[lb][kg]
FlightStateInputs.W_fuel_f = 28400;         % Weight of fuel till end   of cruise
[lb][kg]
% Calculations
RECase = Range_Endurance(Aircraft,FlightStateInputs);
% Display
disp(RECase.ResultsText())
% Plotting
RangeEndurancePlot(Aircraft,FlightStateInputs,0)
%}

%
%====================================
% Steady Climing Flight at (h,V,gamma)
%====================================
% Calculations
ClimbCase = SteadyClimbState(Aircraft,FlightStateInputs);
% Display
disp(ClimbCase.ResultsText())
% Plotting
SteadyClimbPlot(Aircraft,FlightStateInputs,0)
%}
```

Filename: SteadyLevelState.m

```
classdef SteadyLevelState
   properties
       Aircraft
       FlightStateInputs

       %h,V Case
       h
       V
       CL_V
       alpha_V
       delta_e_V
       Thrust_V
       delta_t_V
       Mach_V

       % The V's
       V_max
       V_min
       V_minTC
       V_stall
       Mach_max
       Mach_min

       % aerodynamics
       CL_V_max
       alpha_V_max
       delta_e_V_max
       CL_V_min
       alpha_V_min
       delta_e_V_min
       alpha_stall

       %Thrust
       Thrust_min
       Thrust_max
       delta_t_min
       V_Thrust_min
   end

   methods
       function obj = SteadyLevelState(Aircraft,FlightStateInputs)
            obj.Aircraft = Aircraft;
            obj.FlightStateInputs = FlightStateInputs;

            %make equations easy to read
            obj.h = obj.FlightStateInputs.h;
            obj.V = obj.FlightStateInputs.V;
            A = obj.Aircraft;

            % Steady level flight parameters at V
            [obj.CL_V,obj.alpha_V,obj.delta_e_V,obj.Thrust_V,obj.delta_t_V] =
eqnCLADT_V(A,obj.V,obj.h);
            obj.Mach_V = eqnMach_V(obj.V,obj.h,A.Units);

            % Steady level flight speed
            [obj.V_max,obj.V_min,obj.V_minTC,obj.V_stall]=eqnVVV_h(A,obj.h);
            obj.Mach_max = eqnMach_V(obj.V_max,obj.h,A.Units);
```

```
            obj.Mach_min = eqnMach_V(obj.V_min,obj.h,A.Units);

            [obj.CL_V_max,obj.alpha_V_max,obj.delta_e_V_max,~,~] =
eqnCLADT_V(A,obj.V_max,obj.h);
            [obj.CL_V_min,obj.alpha_V_min,obj.delta_e_V_min,~,~] =
eqnCLADT_V(A,obj.V_min,obj.h);
            [~,obj.alpha_stall,~,~,~] = eqnCLADT_V(A,obj.V_stall,obj.h);

            % Thrust required for steady level flight
            [obj.Thrust_max,obj.Thrust_min,obj.delta_t_min,obj.V_Thrust_min] =
eqnTV_h (A,obj.h);
        end

        function output = ResultsText(obj)
            global t
            output = newline;
            output = sprintf('%s========================================\n',output);
            output = sprintf('%sSteady level flight at H=%g [%s] \n',output,
obj.h,t.Alt);
            output = sprintf('%s========================================\n',output);
            output = sprintf('%s\nParameters at V=%g [%s]\n',output,obj.V,t.V);
            output = sprintf('%s-------------------------------\n',output);
            output = sprintf('%sM @(V=%g)= %g \n',output,obj.V,obj.Mach_V);
            output = sprintf('%sCL @(V=%g) = %g \n',output,obj.V,obj.CL_V);
            output = sprintf('%s%s @(V=%g) = %g
\n',output,t.Alpha,obj.V,obj.alpha_V);
            output = sprintf('%s%se @(V=%g) = %g
\n',output,t.Delta,obj.V,obj.delta_e_V);
            output = sprintf('%sThrust(V=%g)= %g [%s]
\n',output,obj.V,obj.Thrust_V,t.Th);
            output = sprintf('%s%st(V=%g)= %g%%
\n',output,t.Delta,obj.V,obj.delta_t_V);
            output = sprintf('%s\nStall Speed\n',output);
            output = sprintf('%s-------------\n',output);
            output = sprintf('%sV(stall)= %g [%s] \n',output,obj.V_stall,t.V);
            output = sprintf('%s%s(stall)= %g%s
\n',output,t.Alpha,obj.alpha_stall,t.Deg);

            output = sprintf('%s\nMaximum/Minimum Aircraft Velocity\n',output);
            output = sprintf('%s-----------------------------------\n',output);
            output = sprintf('%sV(max)= %g [%s] \n',output,obj.V_max,t.V);
            output = sprintf('%sM(max)= %g \n',output,obj.Mach_max);
            output = sprintf('%sCL @(Vmax) = %g \n',output,obj.CL_V_max);
            output = sprintf('%s%s @(Vmax) = %g \n',output,t.Alpha,obj.alpha_V_max);
            output = sprintf('%s%se @(Vmax) = %g
\n',output,t.Delta,obj.delta_e_V_max);
            output = sprintf('%sV(min)= %g [%s] \n',output,obj.V_min,t.V);
            output = sprintf('%sM(min)= %g \n',output,obj.Mach_min);
            output = sprintf('%sCL @(Vmin) = %g \n',output,obj.CL_V_min);
            output = sprintf('%s%s @(Vmin) = %g \n',output,t.Alpha,obj.alpha_V_min);
            output = sprintf('%s%se @(Vmin) = %g
\n',output,t.Delta,obj.delta_e_V_min);
            output = sprintf('%s\nMaximum/Minimum Thrust\n',output);
            output = sprintf('%s--------------------------\n',output);
            output = sprintf('%sThrust(max)= %g [%s] \n',output,obj.Thrust_max,t.Th);
            output = sprintf('%sThrust(min)= %g [%s] \n',output,obj.Thrust_min,t.Th);
            output = sprintf('%s%st(min)= %g%% \n',output,t.Delta,obj.delta_t_min);
            output = sprintf('%sV @(Thrust_min)= %g [%s]
\n',output,obj.V_Thrust_min,t.V);
        end
    end
end
```

Filename: SteadyLevelPlot.m

```
function SteadyLevelPlot(Aircraft,FlightStateInputs,Type)
    global t

    h = FlightStateInputs.h;
    V = FlightStateInputs.V;
    LevelCase = SteadyLevelState(Aircraft,FlightStateInputs);

    % Flight Envelop
    % ----------------
    % Calculations

    %according to h variation
    h_max = Ceiling(Aircraft).h_max;
    plot_h=linspace(0,h_max,300);
    for iter=1:length(plot_h)
        %iterFSI: iteration for Flight State Inputs
        iterFSI.h = plot_h(iter);
        iterFSI.V = V;
        [V_max_h(iter),V_min_h(iter),V_minTC_h(iter),V_stall_h(iter)] =
```

```
eqnVVV_h(Aircraft,plot_h(iter));
    end

    % According to V variation
    plot_V=linspace(Aircraft.V_plotmin,Aircraft.V_plotmax,500);
    for iter=1:length(plot_V)
        [~,alpha(iter),delta_e(iter),Thrust(iter),delta_t(iter)] =
eqnCLADT_V(Aircraft,plot_V(iter),h);
    end

    % Flight Evelop
    if or(Type == 1,Type == 0)
        if Type == 0; figure(); end
        tmp_fill = fill([min(V_min_h) V_max_h(end:-1:1) V_min_h], [min(plot_h)
plot_h(end:-1:1) plot_h],'y');
        set(tmp_fill,'EdgeColor','none','FaceColor',[0.9 0.9 0.9],'FaceAlpha',0.5);
        xlabel(sprintf('Velocity [%s]',t.V))
        ylabel(sprintf('Altitude [%s]',t.Alt))
        title('Steady Level Flight Envelope')
        hold on;
        plot(V_max_h,plot_h,V_minTC_h,plot_h,V_stall_h,plot_h);
        grid on;
        legend('Envelop','V_{max}','V_{min T}','V_{stall}')
        hold off;
    end

    % Speed Diagrams
    % ----------------
    if or(Type == 2,Type == 0)
        if Type == 0; figure(); end
        plot (plot_V,Thrust)
        xlabel(sprintf('Velocity [%s]',t.V))
        ylabel(sprintf('Thrust required [%s]',t.Th))
        title(sprintf('Thrust required for steady level flight at H=%g [%s]',
h,t.Alt))
        grid on;
        hold on;
        plot_Box(LevelCase.Thrust_max,'Y','T','r','Exceed Thrust_{max}');
        plot_Box(LevelCase.Thrust_min,'Y','B','b','Under Thrust_{min}');
        plot_Box(LevelCase.V_stall,'X','B','k','Below V_{stall}');

line([LevelCase.V_Thrust_min,LevelCase.V_Thrust_min],ylim,'color','g','LineStyle',':'
)
        text(LevelCase.V_Thrust_min*1.01,sum(ylim)*.5,'V @ Thrust_{min}','color','g')
        hold off;
    end
    if or(Type == 3,Type == 0)
        if Type == 0; figure(); end
        plot (plot_V,alpha)
        xlabel(sprintf('Velocity [%s]',t.V))
        ylabel(sprintf('\\alpha%s',t.Deg))
        title(sprintf('AoA (\\alpha) for steady level flight at H=%g [%s]', h,t.Alt))
        hold on;
        plot_Box(LevelCase.alpha_stall,'Y','T','r','Exceed \alpha_{stall}');
        plot_Box(LevelCase.V_stall,'X','B','k','Below V_{stall}');
        grid on;
        hold off;
    end
    if or(Type == 4,Type == 0)
        if Type == 0; figure(); end
        plot (plot_V,delta_e)
        xlabel(sprintf('Velocity [%s]',t.V))
        ylabel(sprintf('\\delta_e%s',t.Deg))
        title(sprintf('Elevator Angle (\\delta_e) for steady level flight at H=%g
[%s]', h,t.Alt))
        grid on;
        hold on
        plot_Box(LevelCase.V_stall,'X','B','k','Below V_{stall}');
        hold off;
    end
    if or(Type == 5,Type == 0)
        if Type == 0; figure(); end
        plot (plot_V,delta_t)
        xlabel(sprintf('Velocity [%s]',t.V))
        ylabel('\delta_t [%]')
        title(sprintf('Throttle (\\delta_t) for steady level flight at H=%g [%s]',
h,t.Alt))
        hold on;
        plot_Box(100,'Y','T','r','Exceed Thrust_{max} (\delta_t > 100%)');
        grid on;
        hold off;
    end
end
```

Filename: eqnVVV_h.m

```
function [V_max,V_min,V_minTC,V_stall] = eqnVVV_h (A,h)
%V_max, V_minTC is due to (TC: Thrust Constrain)

[~,~,rho,~,~,sigma,speed_of_sound] = StdAtm(h,A.Units);

% Stall Speed
V_stall = eqnV_stall_h (A,h);

% Maximum/Minimum aircraft air speed at H
if A.EngineType == 'Jet'
    V_max_roots = roots([0.5*rho*A.S*A.CD0 0 -A.Thrust_max_s*sigma^A.m 0
2*A.K*A.MTOW^2/rho/A.S]); %eq(3.17)
else
    V_max_roots = roots([0.5*rho*A.S*A.CD0 0 0 -A.eta_P*A.Thrust_max_s*sigma^A.m
2*A.K*A.MTOW^2/rho/A.S]); %eq(3.23)
end
V_max = real(max(V_max_roots(V_max_roots>0)));
V_minTC = real(min(V_max_roots(V_max_roots>0)));

%the case V_max is more than the air of speed
if A.sonic < 1
    %case of subsonic
    if V_max >= A.sonic*speed_of_sound
        V_max=A.sonic*speed_of_sound;
    end
    if V_minTC >= A.sonic*speed_of_sound
        V_minTC=A.sonic*speed_of_sound;
    end
end

% V minimum is larger of(V-stall and V-min-TC)
V_min = max(V_minTC,V_stall);
```

Filename: eqnCLADT_V.m

```
function [CL,alpha,delta_e,Thrust,delta_t] = eqnCLADT_V (A,V,h)

[~,~,rho,~,~,~,~] = StdAtm(h,A.Units);
d = 0.5*rho*V^2;
CL = A.MTOW / (d*A.S); %eq(3.1)
alpha = (CL-A.CL0)/A.CL_alpha; %eq(3.14)
delta_e = -(A.CM0+A.CM_alpha*alpha)/A.CM_deltae; %eq(7.12 #ref.1)

if A.EngineType == 'Jet'
    Thrust = d*A.S*A.CD0 + (A.K*A.MTOW^2)/(d*A.S); %eq(3.18)
else %it is propeller
    Thrust = 0.5*rho*V^3*A.S*A.CD0 + (2*A.K*A.MTOW^2)/(rho*V*A.S); %eq(3.24)
end
delta_t = eqnDelta_t(A,Thrust,h);
```

APPENDIX B. ATMOSPHERE MODULE

UPA-Gaziantep has atmosphere calculation module. The module provide results in both; SI and US units systems. The module results are used within the performance equations. To demonstrate the results obtained from atmosphere module, the following diagrams where calculated and plotted.

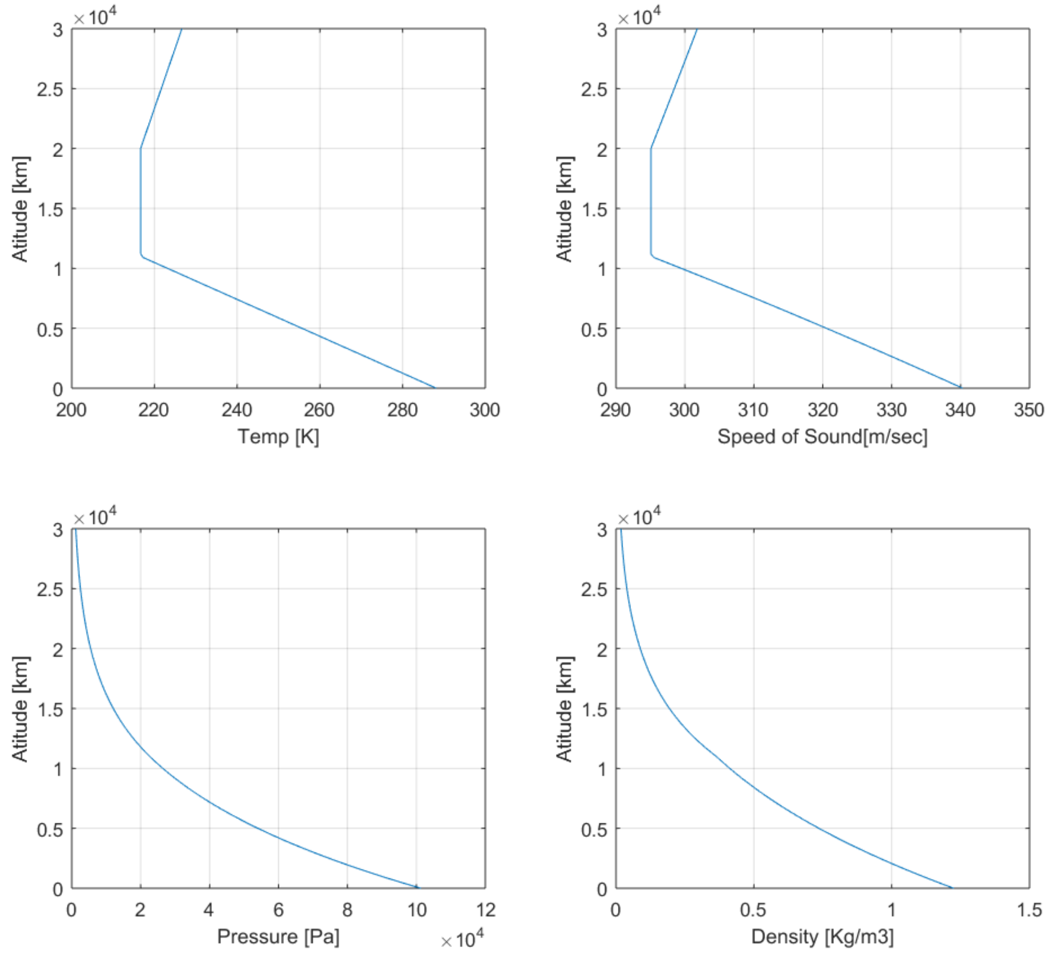


Figure B.1 Standard Atmosphere in SI units

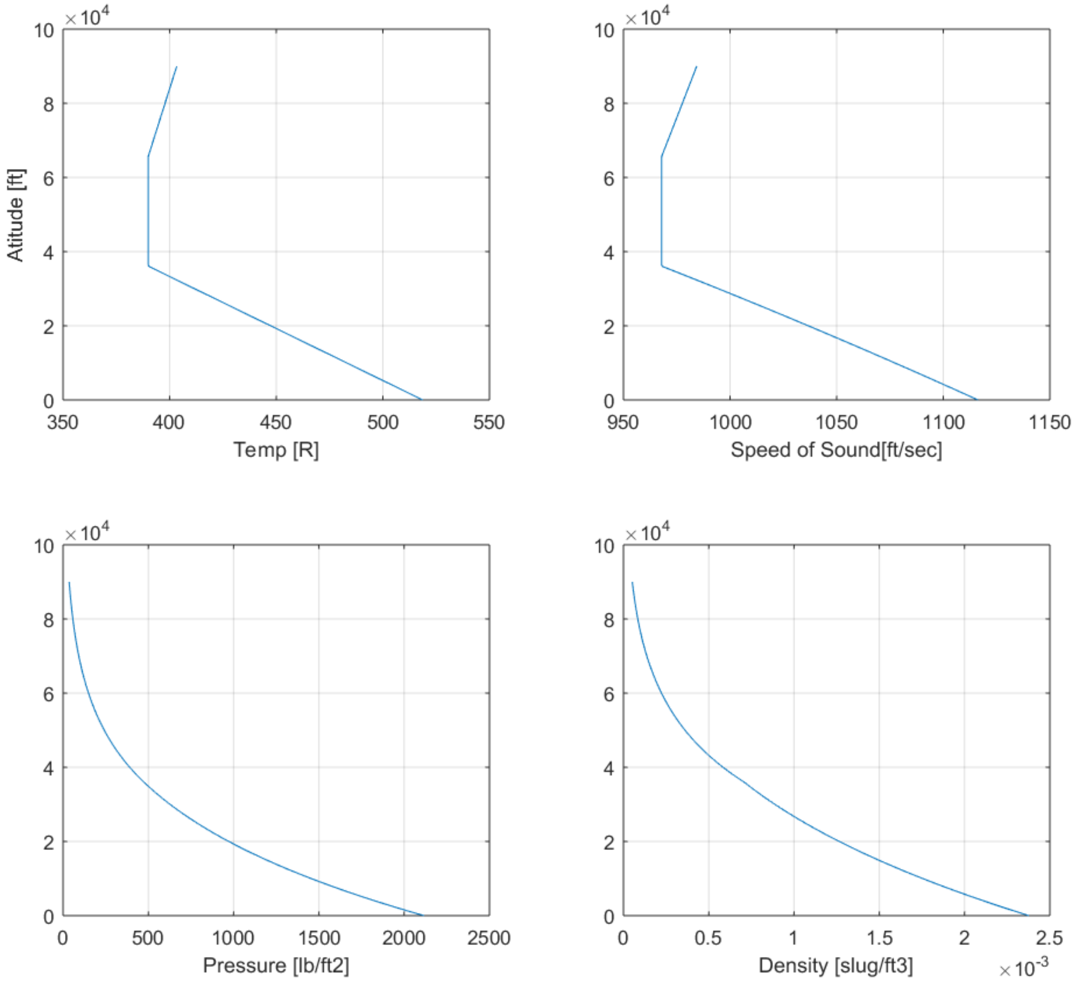
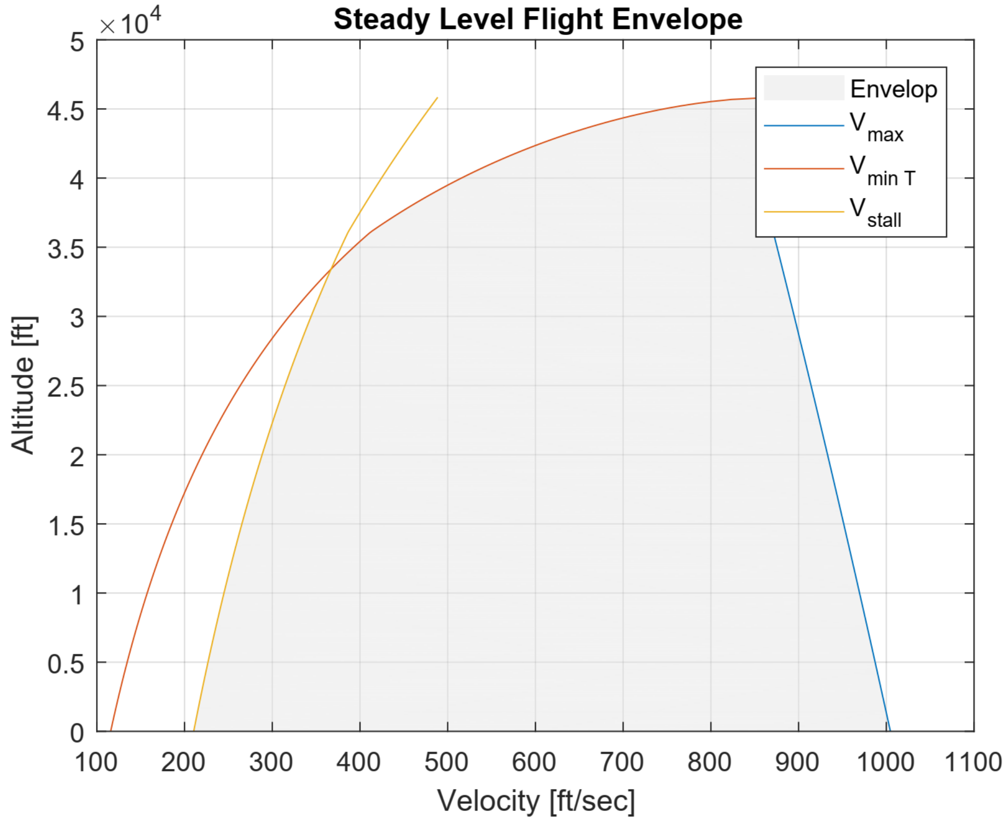


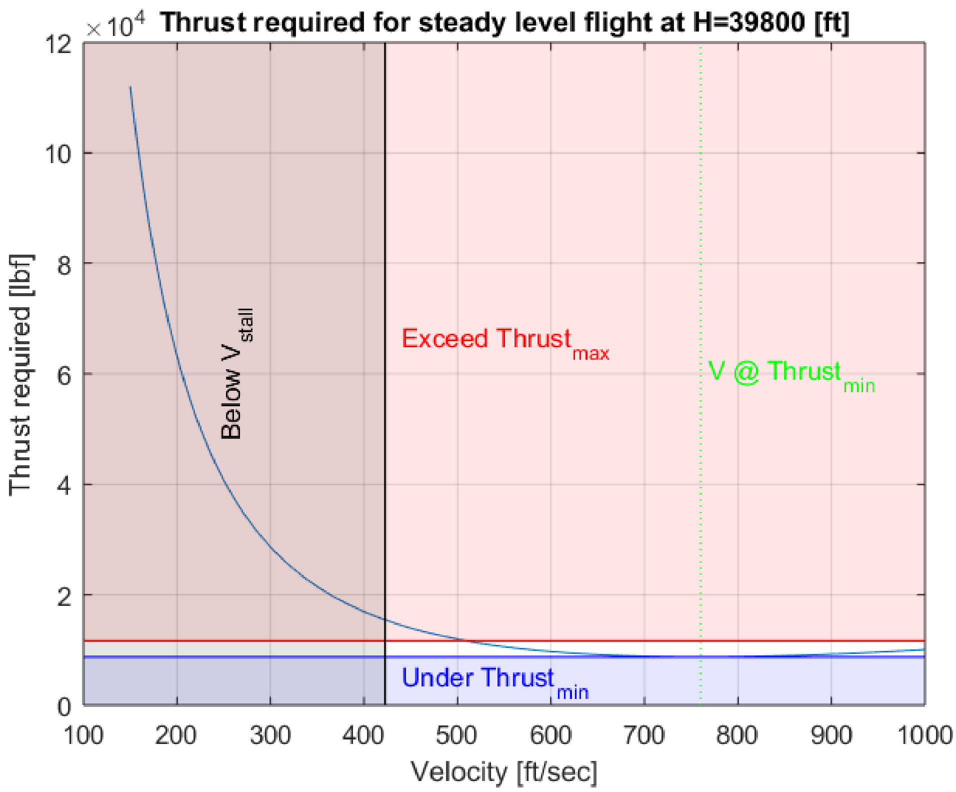
Figure B.2 Standard Atmosphere in US units

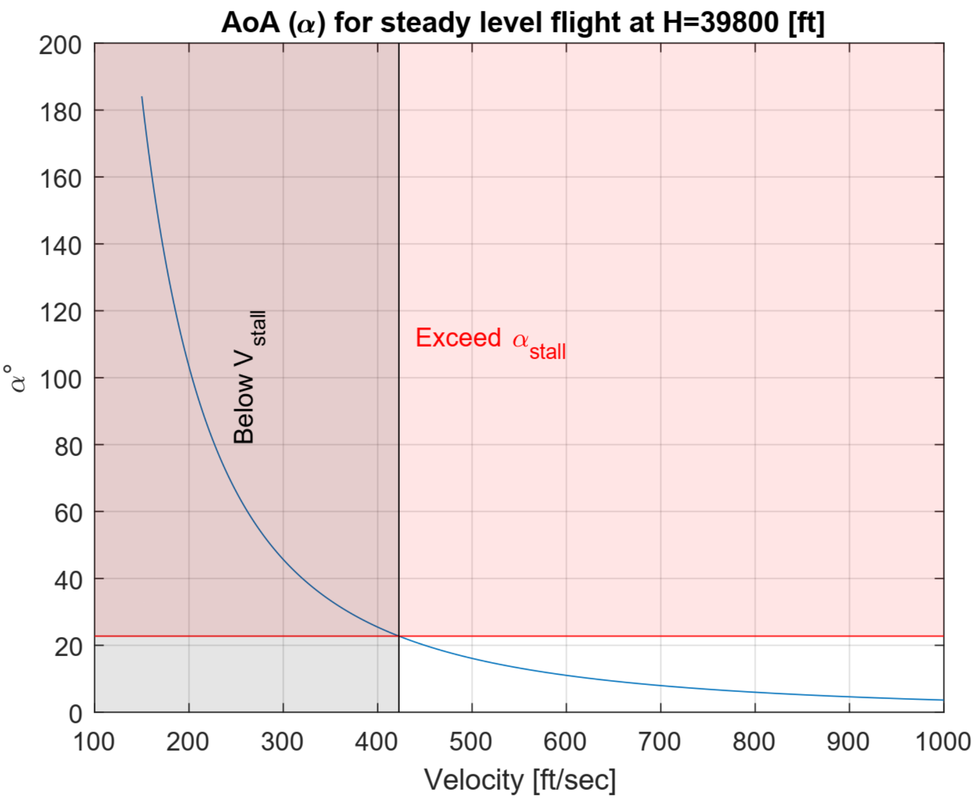
APPENDIX C: SAMPLE RESULTS FROM A320-200 CASE STUDY

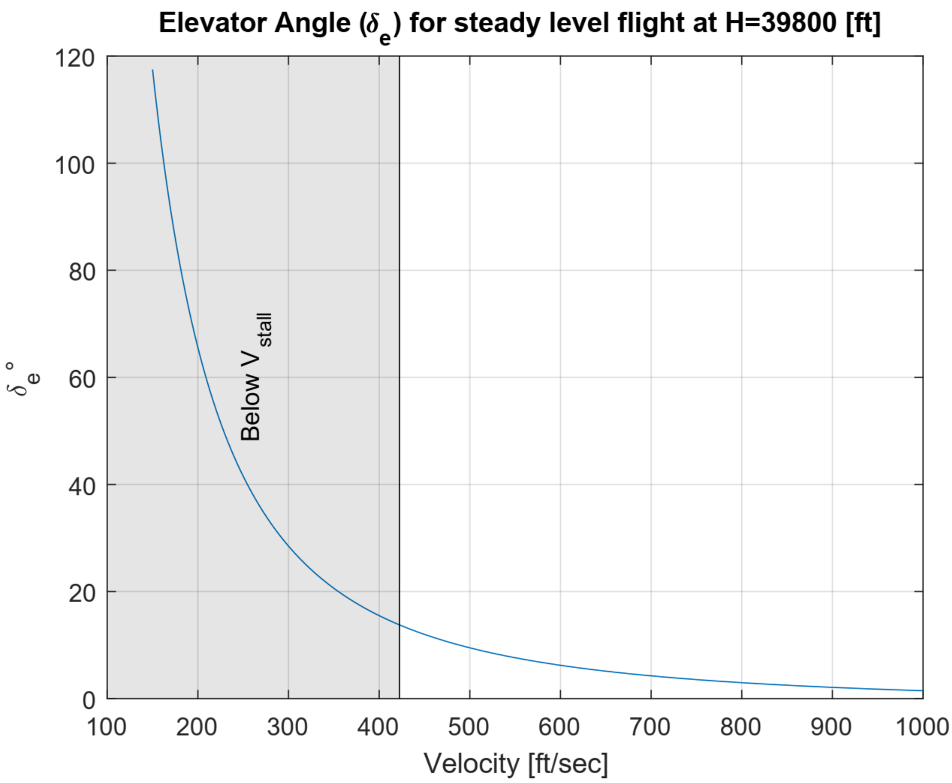
The program produce text results, and plotted results. Below is the results for A320-200 in Steady Level Flight at altitude (h=39,00 ft, V = 0.75 Mach).

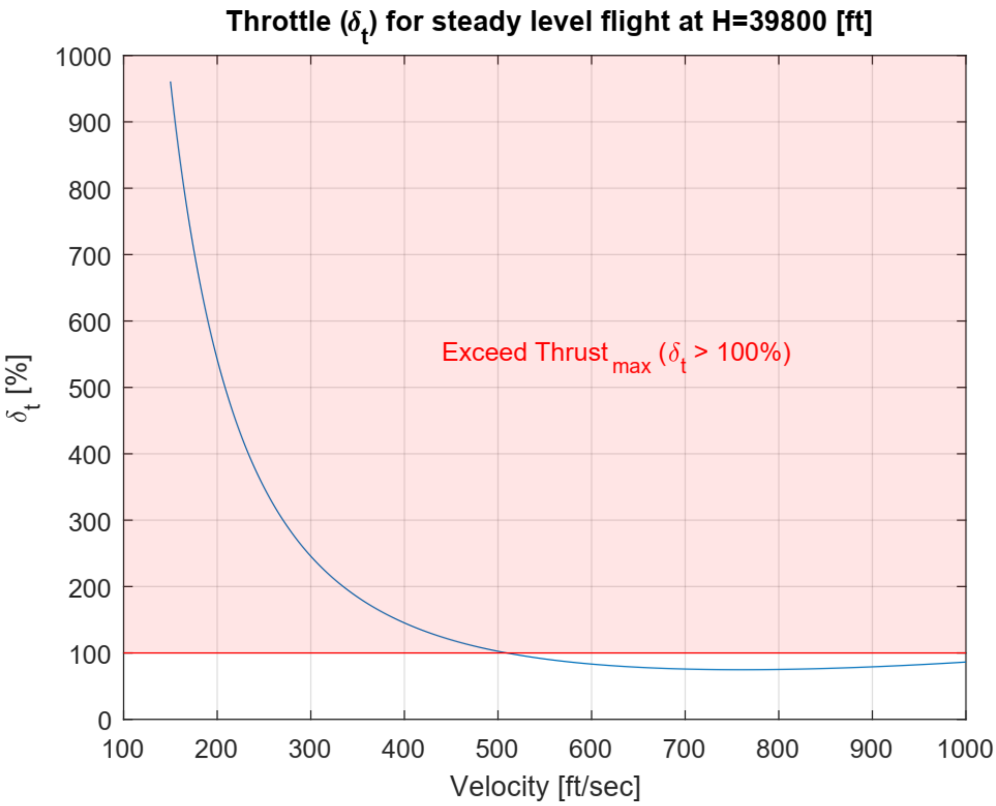
```
=========================================
Steady level flight at H=39800 [ft]
=========================================

Parameters at V=725.911 [ft/sec]
--------------------------------
M @(V=725.911)= 0.75
CL @(V=725.911) = 0.867155
α @(V=725.911) = 7.36712
δe @(V=725.911) = 3.87887
Thrust(V=725.911)= 8755.51 [lbf]
δt(V=725.911)= 75.0926%

Stall Speed
--------------
V(stall)= 422.485 [ft/sec]
α(stall)= 22.7566°

Maximum/Minimum Aircraft Velocity
-------------------------------------
V(max)= 871.093 [ft/sec]
M(max)= 0.9
CL @(Vmax) = 0.602191
α @(Vmax) = 4.95836
δe @(Vmax) = 2.33037
V(min)= 509.373 [ft/sec]
M(min)= 0.526276
CL @(Vmin) = 1.76113
α @(Vmin) = 15.4942
δe @(Vmin) = 9.10339

Maximum/Minimum Thrust
--------------------------
Thrust(max)= 11659.6 [lbf]
Thrust(min)= 8719.15 [lbf]
δt(min)= 74.7808%
V @(Thrust_min)= 577.332 [ft/sec]
```


Steady Level Flight Envelope
×10^4
Altitude [ft]
Velocity [ft/sec]
Envelop
V max
V min T
V stall
5
4.5
4
3.5
3
2.5
2
1.5
1
0.5
0
100
200
300
400
500
600
700
800
900
1000
1100


Thrust required for steady level flight at H=39800 [ft]
×10^4
Thrust required [lbf]
Velocity [ft/sec]
Below V stall
Exceed Thrust max
V @ Thrust min
Under Thrust min
12
10
8
6
4
2
0
100
200
300
400
500
600
700
800
900
1000

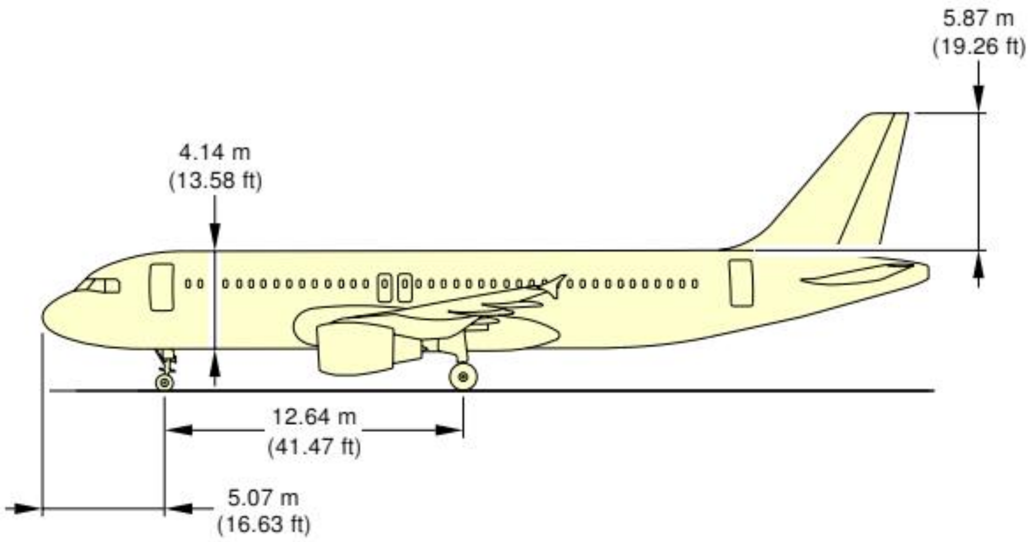

AoA (α) for steady level flight at H=39800 [ft]
Below V_{stall}
Exceed α_{stall}
$\alpha°$
Velocity [ft/sec]
200
180
160
140
120
100
80
60
40
20
0
100
200
300
400
500
600
700
800
900
1000

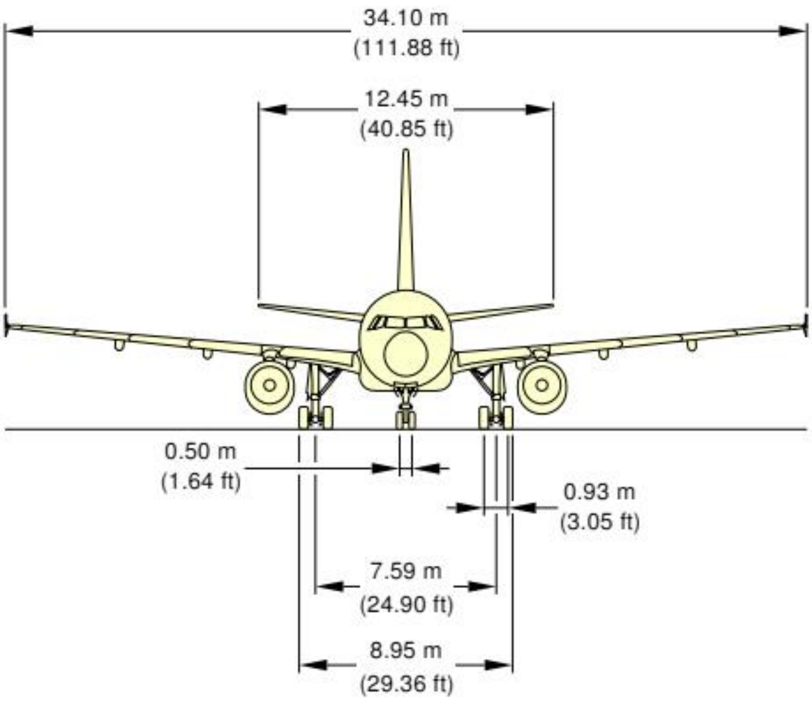

Elevator Angle (δ_e) for steady level flight at H=39800 [ft]
Below V_{stall}
$\delta_e°$
Velocity [ft/sec]
120
100
80
60
40
20
0
100
200
300
400
500
600
700
800
900
1000

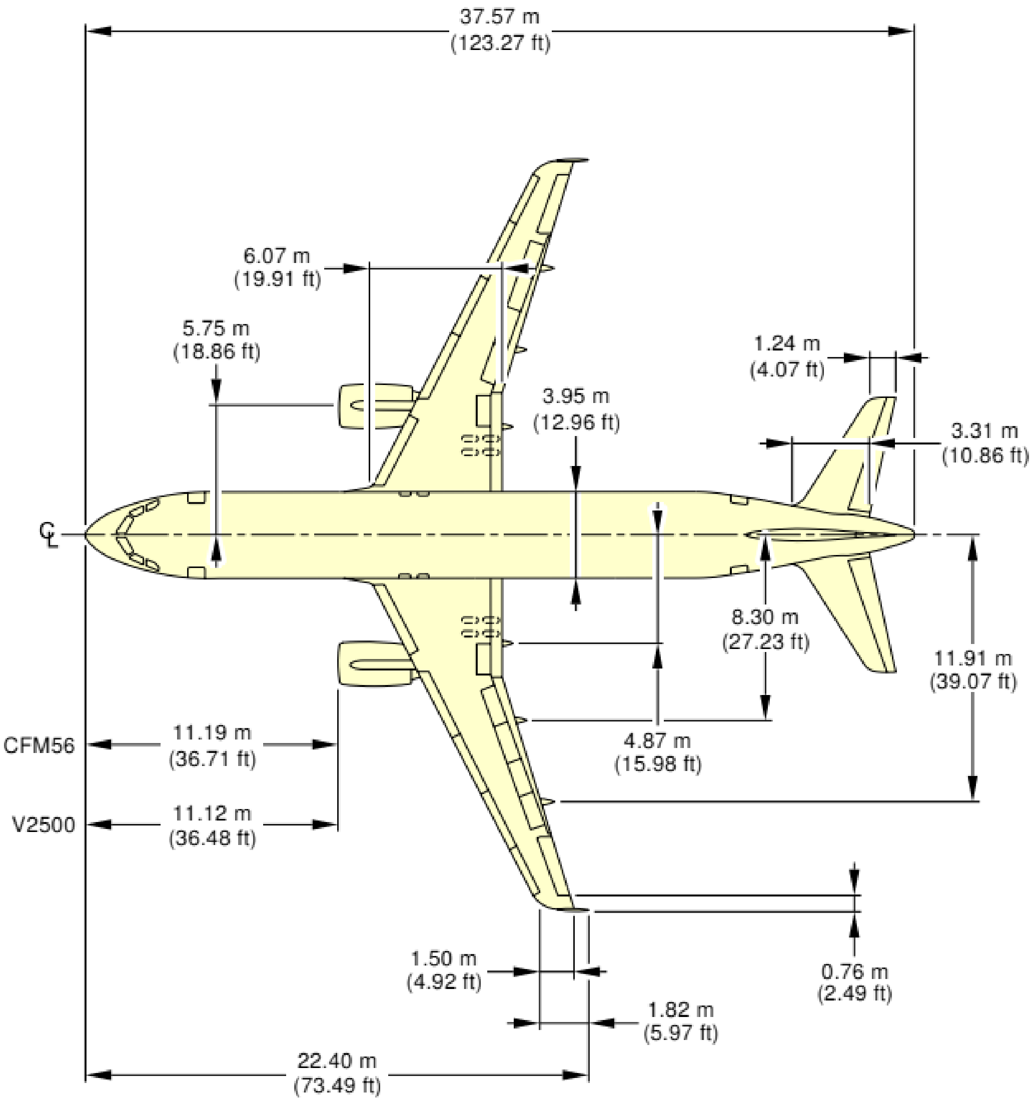

Throttle (δt) for steady level flight at H=39800 [ft]
δt [%]
Velocity [ft/sec]
Exceed Thrust max (δt > 100%)
1000
900
800
700
600
500
400
300
200
100
0
100
200
300
400
500
600
700
800
900
1000

APPENDIX D: AIRBUS AIRCRAFTS INFORMATION

D.1 A320-200 Geometry [50]







D.2 Data Library for A319-100, A320-200, A321-200 [42]

Manufacturer	***AIRBUS***	***AIRBUS***	***AIRBUS***
Type	***A319-***	***A320-***	***A321-***
Model	***100***	***200***	***200***
Initial service date	1995	1988	1993
Engine Manufacturer	CFMI	CFMI	CFMI
Model / Type	CFM56-5A4	CFM56-5A3	CFM56-5B3
No. of engines	2	2	2
Static thrust (kN)	99.7	111.2	142.0
Operational Items:			
Accomodation:			

Max. seats (single class)	153	179	220
Two class seating	124	150	186
Three class seating	-	-	-
No. abreast	6	6	6
Hold volume (m³)	27.00	38.76	51.76
Volume per passenger	0.18	0.22	0.24
Mass (Weight) (kg):			
Ramp	64400	73900	89400
Max. take-off	64000	73500	89000
Max. landing	61000	64500	73500
Zero-fuel	57000	60500	71500
Max. payload	17390	19190	22780
Max. fuel payload	5360	13500	19060
Design payload	11780	14250	17670
Design fuel load	13020	17940	23330
Operational empty	39200	41310	48000
Weight Ratios:			
Ops empty/Max. T/O	0.613	0.562	0.539
Max. payload/Max. T/O	0.272	0.261	0.256
Max. fuel/Max. T/O	0.295	0.256	0.210
Max. landing/Max. T/O	0.953	0.878	0.826
Fuel (litres):			
Standard	23860	23860	23700
Optional			26600

Manufacturer	***AIRBUS***	***AIRBUS***	***AIRBUS***
Type	***A319-***	***A320-***	***A321-***
Model	***100***	***200***	***200***
DIMENSIONS			
Fuselage:			
Length (m)	33.84	37.57	44.51
Height (m)	4.14	4.14	4.14
Width (m)	3.95	3.95	3.95
Finess Ratio	8.57	9.51	11.27
Wing:			
Area (m²)	122.40	122.40	122.40
Span (m)	33.91	33.91	33.91
MAC (m)	4.29	4.29	4.29

Aspect Ratio	9.39	9.39	9.39
Taper Ratio	0.240	0.240	0.240
Average (t/c) %			
1/4 Chord Sweep (°)	25.00	25.00	25.00
High Lift Devices:			
Trailing Edge Flaps Type	F1	F1	F2
Flap Span/Wing Span	0.780	0.780	0.780
Area (m^2)	21.1	21.1	21.1
Leading Edge Flaps Type	slats	slats	slats
Area (m^2)	12.64	12.64	12.64
Vertical Tail:			
Area (m^2)	21.50	21.50	21.50
Height (m)	6.26	6.26	6.26
Aspect Ratio	1.82	1.82	1.82
Taper Ratio	0.303	0.303	0.303
1/4 Chord Sweep (°)	34.00	34.00	34.00
Tail Arm (m)	10.67	12.53	15.20
S_v/S	0.176	0.176	0.176
S_vL_v/S_b	0.055	0.065	0.079
Horizontal Tail:			
Area (m^2)	31.00	31.00	31.00
Span (m)	12.45	12.45	12.45
Aspect Ratio	5.00	5.00	5.00
Taper Ratio	0.256	0.256	0.256
1/4 Chord Sweep (°)	29.00	29.00	29.00
Tail Arm (m)	11.67	13.53	16.20
S_h/S	0.253	0.253	0.253
S_hL_h/S_c	0.689	0.799	0.957
Undercarriage:			
Track (m)	7.60	7.60	7.60
Wheelbase (m)	12.60	12.63	16.90
Turning radius (m)	20.60	21.90	29.00
No. of wheels (nose;main)	2;4	2;4	2;4
Main Wheel diameter (m)	1.143	1.143	1.270
Main Wheel width (m)	0.406	0.406	0.455
Nacelle:			
Length (m)	4.44	4.44	4.44
Max. width (m)	2.37	2.37	2.37
Spanwise location	0.338	0.338	0.338

Manufacturer	*AIRBUS*	*AIRBUS*	*AIRBUS*
Type	***A319-***	***A320-***	***A321-***
Model	***100***	***200***	***200***
PERFORMANCE			
Loadings:			
Max. power load (kg/kN)	320.96	330.49	313.38
Max. wing load (kg/m^2)	522.88	600.49	727.12
Thrust/Weight Ratio	0.3176	0.3084	0.3253
Take-off (m):			
ISA sea level	1750	2180	2000
ISA +20°C SL.	2080	2590	2286
ISA 5000ft	2360	2950	3269
ISA +20°C 5000ft	2870	4390	
Landing (m):			
ISA sea level.	1350	1440	1580
ISA +20°C SL.	1350	1440	1580
ISA 5000ft	1530	1645	1795
ISA +20°C 5000ft	1530	1645	1795
Speeds (kt/Mach):			
V_2	133	143	143
V_{app}	131	134	138
V_{no}/M_{mo}	381/M0.89	350/M0.82	350/M0.82
V_{ne}/M_{me}	350/M0.82	381/M0.89	TBD/M0.89
$C_{L\max}$ (T/O)	2.58	2.56	3.10
$C_{L\max}$ (L/D @ MLM)	2.97	3.00	3.23
Max. cruise :			
Speed (kt)	487	487	487
Altitude (ft)	33000	28000	28000
Fuel consumption (kg/h)	3160	3200	3550
Long range cruise:			
Speed (kt)	446	448	450
Altitude (ft)	37000	37000	37000
Fuel consumption (kg/h)	1980	2100	2100
Range (nm):			
Max. payload	1355	637	1955
Design range	1900	2700	2700
Max. fuel (+ payload)	4158	3672	2602

Ferry range			
Design Parameters:			
$W/SC_{L\mathrm{max}}$	1726.69	1962.27	2211.48
$W/SC_{L\mathrm{toST}}$	2071.39	2423.85	2590.29
Fuel/pax/nm (kg)	0.0553	0.0443	0.0465
Seats x Range (seats.nm)	235600	405000	502200