A New Conceptual Design Tool for General Aviation Aircraft (FLEX)

Ammar Salaymeh
A New Conceptual Design Tool for General Aviation Aircraft (FLEX)

A user-friendly computer implementation of classical design methods

Ammar Salaymeh

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Examiner: Ingo Staack
To my parents
To my wonderful wife
To my supportive family
To your soul Mohammad Rohi Tulimat
To all the refugees in this world
Abstract

This report is a part of a thesis work at the division of Fluid and Mechatronic Systems (Flumes) at Linköping University. The aim of this thesis is to build a robust, advanced, simple, easy to maintain, easy to develop and user-friendly program for the general aviation aircraft conceptual design. The program name is FLEX (Flumes Excel).

The program was developed based on the most famous references in this field such as Raymer [1], Gudmundsson [2] and Torenbeek [3]. Different methods and equations were evaluated to choose the best. In case there is no way to evaluate the methods, they were all implemented and the user has the ability to choose the desired one. Microsoft Excel was chosen to build the program and Excel VBA was used to build macros and functions in order to serve the objectives of the program.

This report explains the used methods, the implementation way, and the program arrangement. It also shows how the results are presented in the program and provides the user with notes about the program using and its limits. The author supposes that the reader is familiar with the basic aerodynamic and aircraft design knowledge and nomenclatures. In order to not extend the report, some methods and theories are referred to the references without deep explanation. In case the used equations were derived by the author, they are explained in detail.
Nomenclature

Abbreviations and Acronyms

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Meaning</th>
</tr>
</thead>
<tbody>
<tr>
<td>AR</td>
<td>Aspect Ratio</td>
</tr>
<tr>
<td>BeX</td>
<td>Berry Excel (aircraft conceptual design tool at LiU)</td>
</tr>
<tr>
<td>$CD_{\alpha}$</td>
<td>Drag coefficient</td>
</tr>
<tr>
<td>CG</td>
<td>Center of Gravity</td>
</tr>
<tr>
<td>CL</td>
<td>Lift Coefficient</td>
</tr>
<tr>
<td>$CL_0$</td>
<td>Lift Coefficient at Zero Angle of Attack</td>
</tr>
<tr>
<td>$CL_{\alpha}$</td>
<td>Lift curve slope</td>
</tr>
<tr>
<td>C-L</td>
<td>Cruise Level</td>
</tr>
<tr>
<td>FAR</td>
<td>Federal Aviation Regulations</td>
</tr>
<tr>
<td>LSA</td>
<td>Light Sport Aircraft</td>
</tr>
<tr>
<td>LiU</td>
<td>Linköping University</td>
</tr>
<tr>
<td>MAC</td>
<td>Mean Aerodynamic Chord</td>
</tr>
<tr>
<td>MGC</td>
<td>Mean Geometric Chord</td>
</tr>
<tr>
<td>Re</td>
<td>Reynolds Number</td>
</tr>
<tr>
<td>SL</td>
<td>Sea Level</td>
</tr>
<tr>
<td>SI</td>
<td>International System of Units</td>
</tr>
<tr>
<td>TR</td>
<td>Throttle Ratio</td>
</tr>
<tr>
<td>T-O</td>
<td>Take-Off</td>
</tr>
<tr>
<td>t/c</td>
<td>Thickness Ratio to The Chord Length</td>
</tr>
<tr>
<td>VBA</td>
<td>Visual Basic for Applications (programming language)</td>
</tr>
</tbody>
</table>

Latin Symbols

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Units</th>
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<tbody>
<tr>
<td>b</td>
<td>Span</td>
<td>[m]</td>
</tr>
<tr>
<td>D</td>
<td>Drag force</td>
<td>[N]</td>
</tr>
<tr>
<td>e</td>
<td>Oswald span efficiency factor</td>
<td>[-]</td>
</tr>
<tr>
<td>g</td>
<td>gravity acceleration</td>
<td>[m/s²]</td>
</tr>
<tr>
<td>J</td>
<td>Propeller advanced ratio</td>
<td>[-]</td>
</tr>
<tr>
<td>L</td>
<td>Lift force</td>
<td>[N]</td>
</tr>
<tr>
<td>M</td>
<td>Mach Number</td>
<td>[-]</td>
</tr>
<tr>
<td>n</td>
<td>Load factor</td>
<td>[-]</td>
</tr>
<tr>
<td>P</td>
<td>Power</td>
<td>[WorKW]</td>
</tr>
<tr>
<td>q</td>
<td>Dynamic Pressure</td>
<td>[Pa]</td>
</tr>
<tr>
<td>R</td>
<td>Range</td>
<td>[m]</td>
</tr>
<tr>
<td>$S_{ref}$ or $S$</td>
<td>Reference area</td>
<td>[m²]</td>
</tr>
<tr>
<td>$S_{Wet}$</td>
<td>Wetted area</td>
<td>[m²]</td>
</tr>
<tr>
<td>T</td>
<td>Thrust force</td>
<td>[N]</td>
</tr>
<tr>
<td>V</td>
<td>Velocity</td>
<td>[m/s]</td>
</tr>
<tr>
<td>Symbol</td>
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<td>Units</td>
</tr>
<tr>
<td>----------</td>
<td>-------------------</td>
<td>--------</td>
</tr>
<tr>
<td>$W_0$</td>
<td>Gross weight</td>
<td>[N]</td>
</tr>
<tr>
<td>$W_e$</td>
<td>Empty weight</td>
<td>[N]</td>
</tr>
<tr>
<td>$W_f$</td>
<td>Fuel weight</td>
<td>[N]</td>
</tr>
<tr>
<td>$W_{crew}$</td>
<td>Crew weight</td>
<td>[N]</td>
</tr>
<tr>
<td>$W_{battery}$</td>
<td>Battery weight</td>
<td>[N]</td>
</tr>
</tbody>
</table>

**Greek Symbols**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
<th>Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\alpha$</td>
<td>Angle of attack</td>
<td>[degree]</td>
</tr>
<tr>
<td>$\beta$</td>
<td>Propeller pitch angle</td>
<td>[degree]</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Air density</td>
<td>[kg/m$^3$]</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Air viscosity</td>
<td>[N.s/m$^2$]</td>
</tr>
<tr>
<td>$\delta_0$</td>
<td>Pressure ratio</td>
<td>[-]</td>
</tr>
<tr>
<td>$\theta_0$</td>
<td>Temperature ratio to the S-L</td>
<td>[-]</td>
</tr>
<tr>
<td>$\Lambda$</td>
<td>Sweep angle</td>
<td>[degree]</td>
</tr>
<tr>
<td>$\alpha_{ZL}$</td>
<td>Zero Lift angle of attack</td>
<td>[degree]</td>
</tr>
</tbody>
</table>

Note: The used units is the SI units, else it will be mentioned.
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1 Introduction

According to Cambridge Aerospace Dictionary [4] and Dictionary of Aviation [5], the general aviation aircraft is a part of civil aviation which includes all facets of aviation except airlines. It encompasses all the aircraft from the glider to the business jet aircraft.

The aircraft must comply with the regulations standards. These regulations are managed and maintain by the Federal Aviation Administration (FAA) in the USA and by European Aviation Safety Agency (EASE) in Europe. Regarding the general aviation aircraft, the used standards are 14 CFR part 23 (FAR-23) in USA and CS-23 in Europe which are similar in most ways, Gudmundsson [2].

The aircraft design process can be divided into several phases. According to Raymer [1], Andersson [6], Sadraey [7], Jenkinson [8] and Torenbeek [9] these phases are.

1. Conceptual design phases.
2. Preliminary design phase.
3. Detail design phase.

Gudmundsson [2] adds two more phases: the requirements phase before the conceptual design phase and the proof-of-concept and testing phase after the detail design phase. The conceptual design is the focus of this thesis.

The designer in the conceptual design phase looks at a number of aircraft concepts, establishes a trade study between these concepts and chooses the best between them for the further study. During this phase the basic characteristics of the selected design layout and the configurations arrangement are determined. Also the basic questions of the performance, the components weights, and the propulsion system are answered.

1.1 Background

The used aircraft conceptual design tool (BeX) at Linköping University is a collection of different semi-empirical and physic-based rules of fidelity Level-0 or Level-1 for conceptual aircraft design, BeX manual [10]. BeX is an advanced tool and gives a very good results. On the other hand, it is a complex and not a user-friendly tool without good documentation. If it is possible to know the theory behinds the used equations and methods, it is difficult to figure out the ways of implementation. These drawbacks lead to that the BeX is not used any more in some courses particularly in TMAL06 course. TMAL06 is the aircraft conceptual design project course at Linköping university according to the study guide [11]. For these reasons, there is a need to build a new aircraft conceptual design tool with another focus.
1.2 The Aim

The aim of the thesis was to develop a new tool for the conceptual design of the general aviation aircraft by using Microsoft Excel. The program must be a user-friendly, robust, easy to maintain and easy to develop. The clear structure and arrangement as well as the nice coloring are required.

1.3 Working Methodology

At the beginning of the thesis, the working method consisted of the following items:

1. In order to understand the aircraft conceptual design process and determine the general working plan, a number of the most famous aircraft design references were reviewed, like Gudmundsson [2], Raymer [1], Sadraey [7], Jenkinson [8], Torenbeek [3] and Kundu [12].

2. reviewing three of existing aircraft design tool, OpenVSP\textsuperscript{1} Bex and sizing-RC-CAT\textsuperscript{2} to figure out the used methods and to get an inspiration.

3. Learning Excel VBA by Watching tutorials and reading articles. Excel VBA allows to control the Excel sheet and to build function and macros.

By the time more references were used since the topics started to be more specific. For instance, Anderson [14] which is an aerodynamic book. Another example, Hepperle [15] and Traub [16] which deal with the electric powered aircraft. When different methods or equations were found, the author tried to determine and choose the best one by evaluating these methods against the real values if it was possible, or choosing the methods that is recommended by the references. After that, the chosen methods were implemented in Microsoft Excel and tested to verify that the implementation way is free from errors and mistakes.

\textsuperscript{1}OpenVSP is a parametric aircraft geometry tool. The predecessors to OpenVSP have been developed by J.R. Gloudemans and others for NASA, official website [13].

\textsuperscript{2}Aircraft conceptual design tool. It was developed by Dr. David Lundstöm.
2 Literature Review


Kundu [12] and Jenkins [8] are mainly considered with commercial aircraft design. The used regulations standards for commercial aircraft are 14 CFR Part 25 and CS-25 regulations which highly vary from the 14 CFR part 23 and CS-23. However some equations and calculations still valid. For instance, the aerodynamic equations and the design process structure are the same for both the commercial and the general aviation aircraft.

Torenbeek [3] is a very good book and worth with information. The book was written to cover both the commercial and the general aviation aircraft. The main drawback is that the book is old and depends on statistics which in turn depend on older data base. Torenbeek wrote a new book [9] Which are easier to follow.

Raymer [11] is almost the most famous aircraft conceptual design book. The book depends mainly on statistical data based on the exist aircrafts. It covers all the design steps and includes numerical examples in almost every section and two example for the entire design process.

Sadraey [7] recommends a design steps for all of the wing, tail, fuselage and control surfaces. These design steps are easy to follow. The book does not include a detailed performance study are stability analysis. Also some used equations and methods are not explained.

Gudmundsson [2] can be considered as a combination of many references and studies regarding the general aviation aircraft design. The book contains very benefit topics not founded in other references and also includes well documented detailed information and equations. The book can be counted as a preliminary design book.

The trend of building the electric aircraft has already begun and in the future, the electric propulsion will become the main used system in the aircraft industry, according to Roland Berger company [17]. Hepperle [15] conducted a study to investigate the potential and the limitations of the electric propulsion system. Both Hepperle [15] and Traub [16] derived simple equations to estimate the electric aircraft range.

Roskam [18], Howe [19], Hale [20] and Anderson [6] are aircraft design books with different approaches. The first three books were used to verify some information and equations. Anderson [6] recommends statistical equations to estimate the aircraft initial weight and the components weights.
Bertin [14] deals with the aerodynamic. The book explained the geometrical parameters effects on the aerodynamic properties and the acted aerodynamic forces on the aircraft. It includes methods to determine the lift and drag coefficient for different wing types. The book gives better understanding for the reason behind the aerodynamic characteristics.

Anderson [21] covers many topics in the flight in facilitated ways. The numerical examples and simplified explanations make the book a good reference to understand the basics.

SAWE.141 paper [22] contains a method to estimate the aircraft components weights. The main target of it is the preliminary design phase. According to Berry [23] the method was used with a method from Torenbeek [24] to estimate the aircraft components weights in BeX.

SAWE.3579 paper [25] includes the component weight estimations methods from different references like Raymer [1] and Torenbeek [4]. The paper states that Raymer method gives the lowest predicted weights comparing to other methods.

SAWE.3692 paper [26] states that Bombardier company developed a physics-based weight estimation method for the conceptual design phase. The paper shows that the method gives minimal errors comparing to the statistical method. The included information are general.

Megson [27] and Peery [28] deal with the aircraft structures. The included information and methods are beyond the conceptual design phase. They can be used in the preliminary phase or even in the detailed design phase.

Mattingly [29] includes a method to estimate the aircraft weight depending on the statistics and the mission profile. It also explains the constraint diagram and present the required equations to plot it.

The required equations and variables to perform the aircraft static and dynamic stability are included in Nelson [30]. The book also contains a method to determine the flight quality.
3 Theory

An aircraft conceptual design tool must be able to do the following tasks:

1. Calculate the initial weight estimation.
2. Create a constraint diagram based on the requirements to determine the critical performance parameters, the thrust to weight ratio and the wing loading.
3. Consider the stall speed characteristics.
4. Based on the selected propulsion methodology, design the propulsion system and determine the available thrust at different flight conditions.
5. Design and study the wing to make sure that the wing suits the mission.
6. Estimate the tail surfaces area and their positions.
7. Design the layout of the fuselage.
8. Determine the total empty weight, the components weights and the fuel weight.
9. Perform a detailed drag analysis of the design.
10. Determine the empty weight CG and the CG envelope.
11. Perform a detailed static and dynamic stability analysis.
12. Perform a detailed performance study (T-O, Climb, Cruise...etc).

Due to time constraint, the last three points were not covered in this thesis. They can be added in the future. However the related theory was reviewed. The author recommends to follow the instructions in chapters (16 - 22) in Gudmundsson [2] to perform the performance study and analysis. These chapters include detailed information and it is easy to follow and understand the used methods. Regarding the aircraft stability, the author recommends to use Nelson [30] as a main reference. To determine the fuselage pitching moment, both Nelson [30] and Gudmundsson [2] in appendix C1 present a way to determine it. The methods are similar in the concept, but the method in Gudmundsson [2] is easier to follow. The dynamic stability study requires many variables to be determined. The equations to determine the variables regarding the control surfaces can be found in chapter (12) at Sadraey [7].

For the rest of the points (first nine points), the related theory is described in the following sections. Some points were combined in one section. For instance, points (2) and (3) are discussed in the sizing section.
3.1 Initial Gross Take off Weight Estimation

Initial weight estimation is usually the first step in the aircraft design process. This estimation can be done depending on the statistic based on the weights of existing aircraft of the same category or depending on the regulations. Raymer [1], Anderson [6], Gudmundsson [2] and Matting [29] suggest methods to estimates the initial weight by using statistics and simplified equations. All these methods follow the same concept. Raymer [1] and Anderson [6] methods are almost the same. The chosen method is Raymer [1] method since it covers more aircraft categories and each category is divided into sub-categories which can enhance the results. For instance, the general aviation aircraft is divided for single engine aircraft and two engines aircraft with two different statistic equations to estimate the empty weight. The chosen method has been extended to include the electric powered aircraft. Equations (1) and (2) are the used equations to estimate the initial weight for non-electric aircrafts and electric aircrafts respectively.

\[
W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - (W_f/W_0) - (W_e/W_0)}
\]  

(1)

\[
W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - (W_e/W_0) - (W_{\text{battery}}/W_0)}
\]  

(2)

Where:

- \(W_{\text{crew}}\) = the crew load in kg or N.
- \(W_{\text{payload}}\) = the payload weight in kg or N.
- \(W_0\) = the estimated gross weight of the aircraft in kg or N
- \(W_f/W_0\) = the fuel fraction.
- \(W_e/W_0\) = the empty weight fraction.
- \(W_{\text{battery}}/W_0\) = the battery weight fraction

3.1.1 Empty Weight Fraction

This term is estimated statistically from existing aircraft. The empty weight can be written as a function of the total weight, as Equation (3).

\[
W_e/W_0 = A \cdot W_0^c
\]  

(3)

Where (A) and (c) are constants. Table 3.1 from Raymer [1] is used to estimate the values for these constants.
3.1.2 Battery Weight Fraction

Equation (4) from Hepperle [15] is chosen to determine the required battery weight depending on the range and the battery energy capacity. Traub [16] suggests another equation to estimate this value but the equation requires more variables to be estimated. In this early step of the design process, it is better to use a simple equation.

\[
W_{battery}/W_0 = \frac{R}{\left( E_{battery} \cdot \eta_{total} \cdot \frac{1}{g} \cdot (\frac{L}{D})_{max} \right)}
\]  

(4)

Where:

- \( R = \) the flight range in m.
- \( \eta_{total} = \) the total efficiency.
- \( (L/D)_{max} = \) the maximum lift over drag ratio. This term will be discussed in detail in the next section.
- \( E_{battery} = \) the battery specific energy capacity.
- \( g = \) the gravity acceleration.

3.1.3 Fuel Fraction

This term depends on many factors like the mission profile, the fuel consumption and the range. To estimate this term, the mission profile must be divided into a number of segments. Then, the weight fraction is estimated for each segment separately. There are usually five types of segments: warm-up, take-off, climb, cruise, loiter, and landing. Table (4) presents historical values of the fuel fraction for the Warm-up, climb and landing segments.

<table>
<thead>
<tr>
<th>Mission Segment</th>
<th>( W_i/W_{i-1} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>Warm-up</td>
<td>0.970</td>
</tr>
<tr>
<td>Climb</td>
<td>0.985</td>
</tr>
<tr>
<td>Landing</td>
<td>0.995</td>
</tr>
</tbody>
</table>

Raymer [1] suggests equations (5) and (6) to estimate the weight fractions for cruise and loiter segments respectively.

\[
W_i/W_{i-1} = \exp \cdot \frac{-R \cdot C}{V \cdot (L/D)}
\]  

(5)

\[
W_i/W_{i-1} = \exp \cdot \frac{-E \cdot C}{(L/D)}
\]  

(6)

Where:

- *note: (i) is the number of segment. For instance, 1 is warm up segment and 2 is take-off segment.
- \( R \) = the range in meter.
- \( C \) = the fuel consumption.
- \( V \) = the velocity at cruise m/s.
- \( E \) = the endurance or loiter time in hour.

The total mission weight fraction is calculated by multiplying the weight fractions of all segments. The fuel weight fraction is related to the total weight fraction as in equation, Raymer \[1\]

\[
\frac{W_f}{W_0} = 1.06 \left(1 - \frac{W_x}{W_0}\right)
\]

(7)

Where
- \( W_x/W_0 \) = the total weight fraction.
- 1.06 = correction factor.

### 3.1.4 Maximum Lift over Drag

Maximum lift over drag value effects directly and mainly the fuel and battery weight estimation. In this early step in the design process, it can’t be calculated. Raymer \[1\] presents the following equation to estimate this term.

\[
\left(\frac{L}{D}\right)_{max} = K_{LD} \cdot \sqrt{\frac{AR}{S_{wet}/S_{ref}}}
\]

(8)

where:
- \( K_{LD} \) = a constant depends on the aircraft type. Table (6) presents the value of this constant based on a statistical data.
- \( AR \) = the estimated aspect ratio.
- \( S_{wet} \) = the estimated wetted area in \( m^2 \).
- \( S_{ref} \) = the estimated reference area \( m^2 \).

*note: the reference \[1\] suggests a statistical way to estimate \( S_{wet}/S_{ref} \). The lift over drag ratio must be corrected before it is used in equations (5) and (6) according to table (5), Raymer \[1\].

<table>
<thead>
<tr>
<th></th>
<th>Cruise</th>
<th>Loiter</th>
</tr>
</thead>
<tbody>
<tr>
<td>Jet</td>
<td>0.866</td>
<td>1</td>
</tr>
<tr>
<td>Prop</td>
<td>1</td>
<td>0.866</td>
</tr>
</tbody>
</table>

Table 5: \((L/D)_{max}\) correction factors, Raymer \[1\]
Table 6: $K_{LD}$ constant for different aircraft types, Raymer [1]

<table>
<thead>
<tr>
<th>Aircraft Type</th>
<th>$K_{LD}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Civil jets</td>
<td>15.5</td>
</tr>
<tr>
<td>Military jets</td>
<td>14</td>
</tr>
<tr>
<td>Retractable prop</td>
<td>11</td>
</tr>
<tr>
<td>Non retractable prop</td>
<td>9</td>
</tr>
<tr>
<td>High aspect-ratio</td>
<td>13</td>
</tr>
<tr>
<td>sailplanes</td>
<td>15</td>
</tr>
</tbody>
</table>

3.2 Initial Sizing

The initial sizing term is used to express the applied process to determine the wing’s area and the needed thrust to fulfill the performance requirements. The constraint and the stall-cruise speeds carpet diagrams are used to complete this mission. Mainly, the constraint diagram sets the values for the thrust and the wing area. The stall-cruise speeds diagram helps to choose the correct point from the constraint diagram and shows the trade-off between the stall speed, cruise speed, and wing area.

3.2.1 Constraint Diagram

The constraint diagram is a powerful method to estimate the wing area and the required thrust. To generate the constraint diagram, the important performance characteristics must be described by using mathematical equations. Every equation must include the thrust-to-weight ratio and weight-to-wing area ratio (the wing loading) or one of them. Gudmundsson [2], Torenbeek [3], and Matting [29] present the required equations, but they differ according to the considered aircraft type (commercial or general aviation) and the followed regulation. For this reason the equations from Gudmundsson [2] are used since the book follow FAR-23 regulation and concentrates on the general aviation aircraft.

Figure (1) presents an example of a constraint diagram from Gudmundsson [2]. The vertical axis presents the thrust (power) to weight ratio, while the horizontal axis presents the wing loading.

The fives curves in Figure (1) present the most important constraints for general aviation aircraft, Gudmundsson [2]. Every constraint will be explained later in this report. There is another constraint which is the landing distance, that not included in this figure. The point located above the curves and has the minimum thrust to weight ratio is typically the design point.

1. **Level constant-velocity turn constraint:**

   Equation (9) is used to determine the required thrust-to-weight ratio to maintain a specific banking load at a specific velocity without losing any altitude, Gudmundsson [2].

   $$\frac{T}{W} = q \cdot \left( \frac{C_{Dmin}}{(W/S)} + k \cdot \left( \frac{n}{q} \right)^2 \cdot \left( \frac{W}{S} \right) \right)$$ (9)
2. Desired rate of climb constraint:
The following mathematical expression is used to determine the required thrust to weight ratio for the desired rate of climb, Gudmundsson [2].

\[
\frac{T}{W} = \frac{V}{V} + \frac{q}{(W/S)} \cdot C_{D_{min}} + \frac{k}{q} \cdot \left( \frac{W}{S} \right) \tag{10}
\]

3. Desired Take-off distance constraint:
It is an important constraint and the following expression is used to calculate the required thrust to weight ratio to maintain the desired ground run distance, Gudmundsson [2].

\[
\frac{T}{W} = \frac{V_{LOF}^2}{2g \cdot S_G} + \frac{q \cdot C_{DTO}}{W/S} + \mu \cdot \left( 1 - \frac{q \cdot C_{LTO}}{W/S} \right) \tag{11}
\]

4. Desired cruise airspeed constraint:
The following mathematical expression is used to calculate the required thrust to weight ratio to achieve the desired cruise speed at a specific altitude, Gudmundsson [2].

\[
\frac{T}{W} = \frac{q}{(W/S)} \cdot C_{D_{min}} + \frac{k}{q} \cdot \left( \frac{W}{S} \right) \tag{12}
\]

5. Desired service ceiling constraint:
The following expression is used to determine the required thrust to weight ratio to achieve the desired service ceiling, Gudmundsson [2].

\[
\frac{T}{W} = \sqrt{\frac{2(V_{ceiling})}{\rho \cdot S \cdot \sqrt{\frac{k}{3 \cdot C_{D_{min}}}}}} + 4 \sqrt{\frac{k \cdot C_{D_{min}}}{3}} \tag{13}
\]

6. Landing distance constraint:
The landing distance constraint will be presented by a vertical line in the
constraint diagram. This is a result of the fact that landing distance is a function only for the wing loading. The following mathematical expression from Raymer [1] is used to determine the required wing loading to achieve a given landing distance.

\[ S_{\text{landing}} = 80 \left( \frac{W}{S} \right) \left( \frac{1}{\sigma \cdot C_{L_{\text{max}}}} \right) + S_a \] (14)

The following list explains the constraints equations terms:

- \( q \) = the dynamic pressure (N). It differs from equation to another depending on the condition.
- \( n \) = the load factor = \( 1 / \cos(\theta) \). where \( \theta \) is the banking angle.
- \( k \) = the lift-induced drag constant.
- \( V_V \) = the vertical speed (m/s).
- \( V_{LOF} \) = the lift-off speed (m/s).
- \( S_G \) = the ground run (m).
- \( C_{L_{TO}} \) and \( C_{D_{TO}} \) = the lift and drag coefficients during take-off run.
- \( \mu \) = the ground friction constant.
- \( V_{V_{\text{ceiling}}} \) = the vertical speed. Typically it is equal to 0.508 m/s according to Gudmundsson [2].
- \( \sigma \) = the density ratio \( \rho_{\text{Landing}} / \rho_{SL} \) between the airport altitude and sea level.
- \( S_a \) = the obstacle-clearance distance. It is equal to 183m for general aviation aircraft according to Raymer [1].

In all the previous constraint equations, the thrust to weight ratio is calculated. But, if the aircraft is a propeller-driven aircraft, the term \( T/W \) (thrust to weight ratio) must be converted to \( P/W \) (power to weight ratio). Equation (15) from Gudmundsson [2] can be used to do this.

\[ T = \frac{\eta_P \cdot P}{V} \] (15)

Where

- \( \eta_P \) = the propeller efficiency.
- \( P \) = the engine power in (Watt).

Both power of the propeller-driven aircraft and the thrust of the jet aircraft will change with the altitude, except the electric powered aircraft. For the jet aircraft, the thrust also changes with the velocity (Mach number). To make the calculated values from the constraints equations comparable and useful, the thrust (power) to weight ratio must be normalized to sea level. The equations from section (2.3) in
this report are used to normalize the power and thrust. It is possible to choose a point from the constraints diagram which meets all previous requirements, but it does not meet the stall speed requirement. To prevent this problem Gudmundsson [2] recommends to plot lines to present selected stalling speeds in the constrains diagram. See Figure [2]. This can be done by plotting the maximum lift coefficients as a function for wing load (W/S) for different stall speeds. Maximum lift coefficient will be presented in the constraint diagram by using a secondary vertical axis. Equation (16) from Gudmundsson [2] is used to do the calculations.

\[
CL_{max} = \frac{1}{q_{stall}} \left( \frac{W}{S} \right)
\]  

(16)

The stall speed limit varies depending on the used regulations. FAR-23 requires the airplane to stall at 61 KCAS (≈ 31 m/s), while the LSA regulations requires the airplane to stall at 45 KCAS (≈ 23 m/s). For ultralight aircraft the limit is 35 KCAS (≈ 18 m/s).

### 3.2.2 Stall Speed - Cruise Speed Carpet Plot

The carpet plot presents the trade-off between the stall speed and the cruise speed for different wing area and how the required maximum lift coefficient is effected. Figure[3] shows an example of the carpet plot from Gudmundsson [2]. This information can help the user to choose the correct wing loading from the constraint diagram.

To generate the carpet plot the stall velocity and the maximum cruise speed for different wing area must be calculated. The stall speed is a function of the maximum lift coefficient and the reference area. Equation[17] are used to calculate the stall speed.
The maximum cruise speed ($V_C$) calculations depend on the propulsion system type. If the aircraft is a propeller-driven aircraft, the following mathematical equation is used, Gudmundsson [2].

$$\rho \cdot S \cdot C_{D_{\text{min}}} \cdot V_C^3 = \eta_P \cdot P_{\text{max}} + \sqrt{(\eta_P \cdot P_{\text{max}})^2 - 4W^2 \cdot V_C^2 \cdot C_{D_{\text{min}}} \cdot k}$$

(18)

If the aircraft is a jet aircraft, the following mathematical expression is used, Gudmundsson [2].

$$V_C = \sqrt{\frac{T_{\text{max}} + \sqrt{T_{\text{max}}^2 - 4C_{D_{\text{min}}}K W^2}}{\rho \cdot S \cdot C_{D_{\text{min}}}}}$$

(19)

where:

- $P_{\text{max}}$ and $T_{\text{max}}$ = The maximum available power and thrust at cruise respectively.

The propeller efficiency is assumed to be constant for different cruise velocities as a simplification. After completing these calculations, a special way will be used to plot the carpet plot. More information about the plotting way can be found at section 3.3.1 at Gudmundsson [2]. In general, the horizontal axis is used to present the cruise speed, while the vertical presents the stall speed. The wing area values plot as vertical lines. The intersection between the desired stall speed and the desired cruise speed gives the required wing area and required maximum lift coefficient.
3.3 Propulsion System

A good propulsion system design and selection are important to the success of any powered aircraft design. Sure, bigger engine means more power and thrust but at the same time, it means more weight and drag. Thrust to weight ratio from the constraint diagram gives a good first step estimation of the required thrust. In later steps, the engine thrust and weight can be optimized. The thrust or power output of the engine determines the aircraft performance and stability. It depends on the altitude and the air velocity (Mach number). So, it is important for any aircraft design software to be able to calculate the available thrust or power at any flight condition.

The propulsion system is not just the engine. It includes other components like propeller, batteries, cables, gearboxes. The detailed study of some of these topics is beyond this thesis objective. However, The propeller is considered in more details since it has a major effect on the generated thrust.

This section is divided into three subsections, available thrust or power, propeller anatomy and engine weight.

3.3.1 Available Thrust or Power

As it is mentioned before, determining the available thrust (power) at any flight condition is a requirement to study the aircraft performance. Both altitude and velocity have an impact on it in different ways depending on the engine type.

There are five possible types of engines.

1. **Piston Engine**

   The power output decreases with altitude increasing. To determine this impact Gagg and Ferrar equation is used as it is recommended at Gudmundsson [2], Equation (20).

   \[ P = P_{SL} \sigma - 0.117 \sigma - 0.132 = P_{SL}(1.123 \sigma - 0.132) \] (20)

   Where
   - \( P_{SL} \) = The engine power at sea level in hp (horsepower).
   - \( \sigma \) = Density ratio.

   This mathematical expressions is used only with normally aspirated engines. The thrust depends on the propeller efficiency and the air velocity.

2. **Turboprops Engine**

   The thrust generated by the turboprop depends on both the altitude and the velocity. It can be calculated by using Equations [21] [22] and [23] according to Gudmundsson [2].

   If \( M =< 0.1 \) then

   \[ T = T_{SL} \delta_0 \] (21)
If \( M > 0.1 \) and \( \theta_0 =< TR \) then
\[
T = T_{SL}\delta_0[1 - 0.96(M - 0.1)^{0.25}] \tag{22}
\]
If \( M > 0.1 \) and \( \theta_0 > TR \) then
\[
T = T_{SL}\delta_0[1 - 0.96(M - 0.1)^{0.25} - \frac{3(\theta_0 - TR)}{8.13(M - 0.1)}] \tag{23}
\]

Where
- \( T \) = Thrust at any condition.
- \( T_{SL} \) = Thrust at sea level.
- \( TR \) = Throttle ratio
- \( \delta_0 \) = Pressure ratio.
- \( M \) = Mach number.
- \( \theta_0 \) = Temperature ratio.

It can be noticed that these equations take the throttle ratio into account. As a form of simplification, the throttle ratio can be assumed to be equal to 1.

3. **Turbojet Engine**

Turbojet engines are used in some general aviation aircraft and UAVs like the Bede BD-5J and the Caproni C22J. Equations \([24]\) and \([25]\) presents how the engine thrust will change with the altitude, Gudmundsson \([2]\).

If \( \theta_0 =< TR \) then
\[
T = T_{SL}\delta_0[1 - 0.3(\theta_0 - 1) - 0.1\sqrt{M}] \tag{24}
\]
If \( \theta_0 > TR \) then
\[
T = T_{SL}\delta_0[1 - 0.3(\theta_0 - 1) - 0.1\sqrt{M} - \frac{1.5(\theta_0 - TR)}{\theta_0}] \tag{25}
\]

4. **Turbofan Engine**

Turbofan engines can be divided into two kinds, low bypass ratio turbofan and high bypass ratio turbofan. The available thrust can be determined by using the equations \([26]\) and \([27]\) for low bypass ratio and equations \([28]\) and \([29]\) for high bypass ratio, Gudmundsson \([2]\).

If \( \theta_0 =< TR \) then
\[
T = T_{SL}\delta_0 \tag{26}
\]
If \( \theta_0 > TR \) then
\[
T = T_{SL}\delta_0[1 - \frac{3.5(\theta_0 - TR)}{\theta_0}] \tag{27}
\]
If \( \theta_0 =< TR \) then
\[
T = T_{SL}\delta_0[1 - 0.49\sqrt{M}] \tag{28}
\]
If \( \theta_0 > TR \) then
\[
T = T_{SL}\delta_0\left[1 - 0.49\sqrt{M} - \frac{3(\theta_0 - TR)}{1.5 + M}\right] \tag{29}
\]
5. **Electric Engine** The available power from the electric engine does not change with altitude. The available thrust depends on the propeller efficiency and the air velocity.

### 3.3.2 Propeller Anatomy

The propeller is a device to convert the mechanical energy from the engine to thrust force. Propeller design and efficiency have a huge effect on the available thrust. This program is not a propeller design program. It focuses on helping the user to choose the right propeller for the aircraft and to determine the propeller efficiency at different flight condition. This section is divided into two subsections, propeller geometry and propeller efficiency.

1. **Propeller Geometry**

   The propeller diameter (D), the pitch angle (β) and the pitch distance (P_D) are considered the most important propeller geometric properties, Figure [4].

   ![Figure 4: Schematic showing the basic propeller geometric properties, Gudmundsson [2]](image)

   The first step in propeller geometry determining is the diameter estimation. Gudmundsson [2] present a method to estimate the propeller diameter depending on its material and on the number of blades, Equations [30] and [31]. Raymer in [1] presents Equation [32] to estimate the pitch distance. Raymer equation is independent of propeller material. So, it can be used to estimate the diameter of the composite propeller.

   **Wooden propeller, Gudmundsson [2]:**

   \[
   D = 10000 \sqrt[4]{\frac{P_{BHP}}{kW \cdot RPM^2 \cdot V_{TAS}}} 
   \]  
   \[(30)\]

   **Metal propeller, Gudmundsson [2]:**

   \[
   D = k_M \cdot \sqrt[4]{P_{PHB}} 
   \]  
   \[(31)\]
generic propeller, Raymer [1]:
\[ D = k_p \cdot \sqrt{P} \] (32)

Where
- \( P_{BHP} \) = Engine power in (hp).
- \( P \) = Engine power in (kW).
- \( k_W, k_M \) and \( k_p \) = Constants depends on the number of blade. See Table 7.
- \( RPM \) = Engine revolutions per minute.
- \( V_{TAS} \) = Cruise speed in (KTAS).

Table 7: The constant of the propeller diameter estimation equations, Raymer [1] and Gudmundsson [2]

<table>
<thead>
<tr>
<th></th>
<th>two blades</th>
<th>three blades</th>
<th>four blades</th>
</tr>
</thead>
<tbody>
<tr>
<td>( K_W )</td>
<td>53.5</td>
<td>75.8</td>
<td>111</td>
</tr>
<tr>
<td>( K_M )</td>
<td>22</td>
<td>18</td>
<td>-</td>
</tr>
<tr>
<td>( K_P )</td>
<td>0.56</td>
<td>0.52</td>
<td>0.49</td>
</tr>
</tbody>
</table>

According to Gudmundsson [2], the propeller diameter is limited by the tip speed. For wooden propeller the tip speed limit is 0.6 Mach, while for metal and composite propellers it is 0.75 - 0.8 Mach. Therefore, it is important to check the tip speed after determining the propeller diameter. The following mathematical expression can be used to calculate the propeller tip speed.

\[ V_{tip} = 2\pi \left( \frac{RPM}{60} \right) \left( \frac{D}{2} \right) \] (33)

The second step is estimating the propeller required pitch distance and calculate the pitch angle. To do that equations 34 and 35 from Gudmundsson [2] are used respectively.

**Required pitch distance:**

\[ P_D \approx 1251 \left( \frac{V_{KTAS}}{RPM} \right) \left( \frac{1}{\eta_P} \right) \] (34)

**Pitch angle:**

\[ tan\beta = \frac{P_D}{2\pi \cdot r_{ref}} \] (35)

Where:
- \( r_{ref} \) = the reference radius, usually 78\% of the propeller radius.
- \( V_{KTAS} \) = the airspeed in KTAS (knot).
2. **Propeller Efficiency**

Raymer [1], Gudmundsson [2] and Roskam [18] cover this topic. Two ways to define the propeller efficiency were chosen. The first one is the Rankine-Froude momentum theorem or actuator disc theory. The theorem is based on some assumptions and simplifications. One of the assumptions is that the propeller is replaced by an infinitesimally thin actuator disc. Another one is that the flow is inviscid (no drag, no momentum diffusion) and incompressible, Gudmundsson [2]. More information and explanation can be found at the reference. This method can be used only to estimate the efficiency for the constant speed propellers.

The second way to estimate the propeller efficiency assumes that the user knows the propeller’s manufacturer data. The propeller properties ($C_P$, $C_T$, $\eta_P$) formulations are written as functions of the advance ratio ($J$). Equations (36), (37) and (38) show an example. Equation (39) is used to calculate the advanced ratio. All these equations from Gudmundsson [2].

\[
\eta_P = 0.096574 + 1.703049J - 0.952281J^2 \tag{36}
\]

\[
C_T = 0.162133 - 0.106480J - 0.038208J^2 \tag{37}
\]

\[
C_P = 0.058005 + 0.084893J - 0.120439J^2 \tag{38}
\]

\[
J = \frac{60 \cdot V}{RPM \cdot D} \tag{39}
\]

After determining the propeller efficiency, the available thrust can be calculated by using Equation (15).

3.3.3 **Engine Weight**

It is possible to estimate the piston engine and turbofan engine weight by using Equation (40) and Equation (41). These two equations are statistical equation from Gudmundsson [2].

\[
W_{ENG} = \left( \frac{(P/745.7) - 21.55}{0.5515} \right) \times 4.448 \tag{40}
\]

\[
W_{ENG} = \left( \frac{P \times 0.224809 - 153.6}{5.646} \right) \times 4.448 \tag{41}
\]
3.4 Wing

The wing is the most critical and important step in the aircraft layout design. The primary function of the wing is to generate enough lift. But also other functions must be considered like minimizing the drag, pitching moment and weight.

3.4.1 Layout and Equivalent trapezoidal wing

Most of the general aviation aircraft have one of the following three wing configurations.

1. Trapezoidal Wing:

   It is the simplest geometric shape, Figure (5). This kind of wings can be described by using four Inputs the reference area ($S$), the aspect ratio ($AR$), the taper ratio ($\lambda$) and the leading edge sweep angle ($\Lambda_{LE}$). The other geometric parameters can be calculated by using the following mathematical expressions from Gudmundsson [2].

   ![Figure 5: Trapezoidal wing shape from Gudmundsson [2] with modification](image)

   Wing span:
   \[ b = \sqrt{AR \cdot S} \]  

   (42)

   Root chord:
   \[ C_r = \frac{2 \cdot b}{(1 + \lambda) \cdot AR} \]  

   (43)

   Tip chord:
   \[ C_t = C_r \cdot \lambda \]  

   (44)

   Average chord:
   \[ C_{avg} = \frac{C_r + C_t}{2} = \frac{C_r}{2} \cdot (1 + \lambda) \]  

   (45)

   Mean geometric chord:
   \[ C_{MGC} = MGC = \left(\frac{2}{3}\right) \cdot C_r \cdot \left(\frac{1 + \lambda + \lambda^2}{1 + \lambda}\right) \]  

   (46)

   Y location of MGC:
   \[ Y_{MGC} = \left(\frac{b}{6}\right) \cdot \left(\frac{1 + 2\lambda}{1 + \lambda}\right) \]  

   (47)
X location of MGC:

\[ X_{MGC} = Y_{MGC} \cdot \tan(\hat{LE}) \]  \hspace{1cm} (48)

Mean aerodynamic chord:
It is assumed to be equal to Mean geometric chord.

2. Compound wing:

Compound wing means that the wing consists of different segments, Figure [6]. It is possible to design wide ranges of wings even elliptical ones by using this option.

![Figure 6: Compound wing shape from Gudmundsson [2] with modification](image)

3. Semi elliptical wing:

To design this kind of wing, the leading edge must be described by a mathematical equation or by points. Root chord and span are simply just lines, Figure [7].

![Figure 7: Semi wing shape from Gudmundsson [2] with modification](image)

At a conceptual design phase, it is possible and acceptable to use the trapezoidal equivalent wing (if the wing is semi elliptical or compound) to simplify the study. Figure [8] presents an example for an equivalent trapezoidal wing of a compound wing. To get the equivalent trapezoidal wing the original wing must be divided into a number of segments (N). More segments give more accurate results. Then the following mathematic expressions from Gudmundsson [2] are used:

Segment area:

\[ S_i = y_i \cdot \left( \frac{c_i + c_{i+1}}{2} \right) \]  \hspace{1cm} (49)

where:

- \( y_i \) = the segment span (m).
- \( c \) = the chord (m).
• $i$ = the segment number.

Original wing area:

$$S_0 = 2 \cdot \sum_{i=1}^{N-1} S_i$$ (50)

Weighted wing area:

$$S_W = \frac{b_0}{S_0} \cdot \left( \sum_{i=1}^{N-1} c_i \cdot S_i + \sum_{i=1}^{N-1} c_{i+1} \cdot S_i \right)$$ (51)

where:

• $b_0$ = the original wing span (m).

Equivalent root chord:

$$C_{RE} = \frac{2}{S_W} \sum_{i=1}^{N-1} c_i \cdot S_i$$ (52)

Equivalent Tip chord:

$$C_{TE} = \frac{2}{S_W} \sum_{i=1}^{N-1} c_{i+1} \cdot S_i$$ (53)

Equivalent leading edge sweep angle:

$$\Lambda_{LE} = \frac{2}{S_0} \sum_{i=1}^{N-1} \Lambda_i \cdot S_i$$ (54)

Figure 8: Equivalent wing for a compound wing, Gudmundsson [2]
### 3.4.2 Aerodynamic Properties

The considered aerodynamic properties is the Oswald span efficiency factor, lift coefficient, the moment coefficient, the maximum lift coefficient and the stall speed.

1. **Oswald span efficiency:**

   Several methods to estimate the Oswald span efficiency factor can be found in different aircraft design books. The following equations present different estimation methods from Gudmundsson [2] and Raymer [1].

   **Raymer estimation for straight wing, Raymer [1]:**
   
   \[
   e = 1.78(1 - 0.045AR^{0.68}) - 0.64 \tag{55}
   \]

   **Raymer estimation for swept wing, [1]:**
   
   \[
   e = 4.61(1 - 0.045AR^{0.68})(\cos \angle_{LE})^{0.15} - 0.64 \tag{56}
   \]

   **Brandt estimation, Gudmundsson [2]:**
   
   \[
   e = \frac{2}{2 - AR + \sqrt{4 + AR^2(1 + \tan^2\angle_{tmax})}} \tag{57}
   \]

   where:
   
   - \(\angle_{tmax}\) = sweep angle at maximum wing thickness line (degree).
   - \(\angle_{LE}\) = sweep angle at leading edge (degree).

   **Douglas estimation, Gudmundsson [2]:**
   
   \[
   e = \frac{1}{\pi . AR . r . C_{D_{min}} + 1/((1 + 0.03t - 2t^2)u)} \tag{58}
   \]

   where:
   
   - \(t\) = the fuselage width to wingspan ratio = \(w_{fus}/b\).
   - \(w_{fus}\) = the maximum width of the fuselage.
   - \(u\) = the correction factor for nonelectrical wing. Typically 0.98 to 1.
   - \(r\) = parasitic correction factor \(r = 0.38 - \angle_{LE}/3000 + \angle_{LE}^2/15000\)

   **Lifting line theory estimation, Gudmundsson [2]:**
   
   \[
   e = 1/(1 + \delta) \tag{59}
   \]

   Where:
   
   - \(\delta\) = the induced drag factor. It can be obtained either by implementing the lift line theory or from Figure [9]. This method can not be used if the aspect ratio is less than four.
2. Wing Lift Coefficient

The lift coefficient curve for the wing or any 3D lifting surface differs from the lift coefficient curve for its airfoil. Figure (10) presents the lift curves for an elliptical wing and its airfoil. It is clear that the wing must operate at a larger angle of attack (AOA) to generate the same lift coefficient. This fact cannot be ignored. It can also be observed that the zero lift angle of attack is the same for both curves. To estimate the lift coefficient there are many equations and methods at different references. Bertin [14], Nelson [30] and Gudmundsson [2] present Equation (60) to calculate the lift curve slope.

\[
CL_\alpha = \frac{C_{l\alpha}}{1 + C_{l\alpha}/(\pi \cdot AR)}
\]  

Gudmundsson [2] and Sadraey [7] present Equation (60) to do the same mission, but Sadraey simplified the equation. The same references present the lifting line theory to calculate the lift coefficient for a 3D lifting surface. The lifting line theory was chosen since Gudmundsson [2] presents all the previous method and recommends the lifting line theory.

The lifting line theory assumes that the wing is straight (not swept). It does not take into account the dihedral angle. But it considers the twisting angle and the changing of the chord and the airfoil along the span. This method can be used to calculate the lift curve slope for the wings whose aspect ratio (AR) is bigger than four.

This method mathematically replaces the wing with a number of constant strength vortices which are referred to with the Greek letter \( \Gamma \) in Figure (11). Equation (61) must be solved to determine the matrix A which is the heart of this theory, Sadraey [7].
\[ \mu (\alpha_{ZL} - \alpha) = \sum_{n=1}^{N} A_n \cdot \sin(N\theta) \cdot \left(1 + \frac{\mu \cdot N}{\sin(\theta)}\right) \] (61)

Where:

- \( N \) = number of segments (number of vortices).
- \( \mu \) = dimensionless parameter. \( \mu = C_i \cdot Cl_\alpha / 4b \).
- \( \alpha_{ZL} \) = the zero lift angle of attack.
- \( \alpha \) = segment’s angle of attack.
- \( \theta \) = the corresponding angle to each segment figure \[12\].
- \( C_i \) = the segment’s mean aerodynamic chord.
- \( Cl_\alpha \) = the segment’s lift curve slope.
- \( b \) = wing span.
The lift coefficient can be calculated by using equation (62). More details can be found at chapter (9) in Gudmundsson [2] and chapter (5) in Sadraey [7].

\[ CL = \pi \cdot AR \cdot A_1 \]  

(62)

As mentioned before, Gudmundsson [2] and Sadraey [7] include the lifting line theory. Gudmundsson [2] considers neither the wing setting angle nor the wing twist angle which have a huge effect on the generated lift. But it presents a way to calculate the lift curve slope. This way requires the user to provide the angle of attack. Sadraey [7] considers both wing setting angle and the twisting angle, but it does not present any way to calculate the lift curve slope. The informations from both references are used to build the lifting line theory code in the program. The main challenge was to find a way to calculate the wing angle of attack which takes the twisting angle into the consideration.

The lift at zero angle of attack is equal to the lift curve slope times the zero lift angle of attack, Equation (63).

\[ CL_0 = C_{L\alpha} \times \alpha_{ZL} \]  

(63)

where:

- \( C_{L\alpha} \) = Wing lift curve slope.
- \( \alpha_{ZL} \) = The zero lift angle of attack.

This fact allows to write the Lift curve slope equation as the following mathematical expression.

\[ CL = C_{L\alpha}(\alpha + \alpha_{ZL}) \]  

(64)

Where

- \( \alpha \) = Angle of attack.
The lift line theory is used to calculate the lift coefficient value $CL_1$ at cruise condition. It can be written as in Equation (65).

$$CL_1 = CL_\alpha (\alpha_{cruise} + \alpha_{ZL}) \quad (65)$$

By increasing each segment’s angle of attack by one degree and implementing the lifting line theory again, the new lift coefficient value $CL_2$ is calculated. It can be calculated as in Equation (66).

$$CL_2 = CL_\alpha (\alpha_{cruise} + \alpha_{ZL} + 1) \quad (66)$$

Equations (65) and (66) which include two unknown variables ($\alpha_{cruise}$ and $CL_\alpha$) allow to derive Equation (67). This equation is used to calculate the angle of attack at cruise.

$$\alpha_{cruise} = \frac{CL_1 \times (\alpha_{ZL} + 1) - CL_2 \times \alpha_{ZL}}{CL_2 - CL_1} \quad (67)$$

After calculating the angle of attack at cruise, Equations 65 and 66 are used to calculate the lift curve slope and the lift coefficient at zero angle of attack respectively.

3. **Wing Maximum Lift Coefficient and Stall Angle**

The maximum lift coefficient affects the wing stall characteristics. So, it is important to calculate the $CL_{max}$ of the wing. To estimate this term, there are different methods. Two of those methods were chosen. The first method is called rapid estimation method from Gudmundsson [2]. As the name suggests, it is a rapid and simplified method. This method works well for the straight wings (Sweep-angle = 0) or with a low sweep angle. But, for high swept wings, it gives low predict. The second method is USAF DATCOM method from Gudmundsson [2]. This method is more complex and requires extracting some constants from figures. But it gives a good estimation for both straight and swept wing. This method is applicable for only high aspect-ratio wings. Both methods can be divided into several steps.

- **Rapid $CL_{max}$ Estimation:**
  
  **Step 1:** Calculate the $Cl_{max}$ for the airfoil by using the low of effectiveness, Equation (68).
  
  $$Cl_{max} = (Cl_{max})_{Root} + \frac{2y_{MGC}}{b} \left[ (Cl_{max})_{Tip} - (Cl_{max})_{Root} \right] \quad (68)$$

  **Step 2:** Calculate the $CL_{max}$ for the wing from the following equation.
  
  $$CL_{max} = 0.9 \times Cl_{max} \quad (69)$$

  **Step 3:** Correct the value for the swept wing by multiplying the result with correction factor $K_\Lambda$.
  
  $$K_\Lambda = \cos^3 \Lambda_{LE} \quad (70)$$
USAF DATCOM method

Step 1: Determine the taper ratio correction factor.

\[ C_1 = 0.5 \sin \left( \pi (1 - \lambda)^{1.5} + 0.8 \sin^{0.4} (\pi (1 - \lambda)^2) \right) \]  
(71)

Step 2: Determine if the wing fulfill the methods application conditions. The mathematical expression of this condition is the following.

\[ AR > \frac{4}{(C_1 + 1) \cos \Lambda_{LE}} \]  
(72)

Step 3: Determine the leading edge parameter from equation

\[ \Delta y = k \frac{t}{c} \]  
(73)

where: \((k)\) is a constant that depends on the airfoil shape, and \(t/c\) is the airfoil thickness.

Step 4: Determine the maximum lift ratio \((C_{L_{\text{max}}}/C_{l_{\text{max}}})\) from Figure 13.

Step 5: Determine the Mach numbers correction factor \((\Delta C_{L_{\text{max}}})\) from Figure 14. Gudmundsson [2] presents the Mach correction factor for 0 and 20 degree leading edge sweep angle.

Step 6: Calculate the wing maximum lift coefficient \(C_{L_{\text{max}}}\) by using the following equation.

\[ C_{L_{\text{max}}} = \left( \frac{C_{L_{\text{max}}}}{C_{l_{\text{max}}}} \right) C_{l_{\text{max}}} + \Delta C_{L_{\text{max}}} \]  
(74)
After calculating $CL_{\text{max}}$ by using any of the previous methods. The wing stall angle can be calculated by using one of the two Equations (75) and (76).

\[ \alpha_{\text{stall}} = \frac{CL_{\text{max}} - CL_0}{CL_{\alpha}} \]  
\[ (75) \]

\[ \alpha_{\text{stall}} = \frac{CL_{\text{max}}}{CL_{\alpha}} + \alpha_{ZL} + \Delta \alpha_{\text{stall}} \]  
\[ (76) \]

More details about Equation (76) can be found in Gudmundsson [2].

4. **Wing Pitching Moment Coefficient**

To compute the wing pitching moment equation (77) from reference [2] is used:

\[ (C_{ma})_{3D} = C_{L\alpha} \cdot \left( \frac{C_{ma}}{CL_{\alpha}} \right) \]  
\[ (77) \]

where: $C_{ma}$ = the airfoil pitching moment slope.

3.4.3 **Wetted Area and Volume**

It is assumed that the used airfoil at the root and tip chords are the same airfoils. Gudmundsson [2] presents the following mathematical expressions to calculate the wing volume and its wetted area. These equations were modified by the author.

**Total wing volume:**

\[ V = \frac{b \cdot C_r^2}{12} \left( (1 + \lambda^2) \cdot (k + 3) \cdot t/c \right) \]  
\[ (78) \]

where:

- $C_r$ = the wing root chord.
• \( k \) = the maximum thickness airfoil location.
• \( t/c \) = the thickness to chord ratio.

**Wing wetted area without correction:**

\[
S_{\text{wet}} = \frac{b \cdot C_r}{4} \left( 1 + \lambda \right) \left[ \sqrt{(t/c)^2 + 16k^2} + 2\sqrt{(t/c)^2 + 4(1-k)^2} \right] 
\]

(79)

**Wing wetted area with correction**

\[
S_{\text{wet}} = \frac{(b - D) \cdot C_r}{4} \left( 1 + \lambda \right) \left[ \sqrt{(t/c)^2 + 16k^2} + 2\sqrt{(t/c)^2 + 4(1-k)^2} \right] 
\]

(80)

Where \( D \) is the fuselage average width in the section where it is connected to the wing. Note that the root chord is not the same. See Figure [15]. This means also that the taper ratio will change.

![Figure 15: The root chord with considering the fuselage (right) and without (left), [2](#)](image)

### 3.4.4 Lift Distribution Over The Wing

Implementing the lift line theory allows to determine the lift coefficient distribution over the wing. This information can be useful in the farther design study, specially for the structure study. Figure [16](#) presents an example of the lift coefficient distribution over an elliptical wing.

![Figure 16: Lift coefficient distribution over the wing, Sadraey [7](#)](image)
3.5 Tail

The tail has many configurations and it is impossible to cover all of them in the program. The chosen configurations to be covered were four configurations which are the conventional tail, T-tail, V-tail and cruciform tail, Figure (17). To ease the description, this section is divided into three subsections.

![Tail configurations](image)

Figure 17: The covered tail configurations in the program, Raymer [1]

3.5.1 Geometry

To design the tail, the tail volume coefficient and the tail arm must be determined. Table 8 from Raymer [1] can be used to determine the first tail coefficients estimation.

<table>
<thead>
<tr>
<th>Aircraft category</th>
<th>Horizontal stabilizer volume</th>
<th>Vertical stabilizer volume</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sailplanes</td>
<td>0.05</td>
<td>0.02</td>
</tr>
<tr>
<td>Homebuilt</td>
<td>0.05</td>
<td>0.04</td>
</tr>
<tr>
<td>General aviation one engine</td>
<td>0.07</td>
<td>0.04</td>
</tr>
<tr>
<td>General aviation Twin engine</td>
<td>0.08</td>
<td>0.07</td>
</tr>
<tr>
<td>Agricultural</td>
<td>0.05</td>
<td>0.04</td>
</tr>
<tr>
<td>Twin turboprop</td>
<td>0.9</td>
<td>0.08</td>
</tr>
<tr>
<td>Flying boat</td>
<td>0.7</td>
<td>0.06</td>
</tr>
<tr>
<td>Jet trainer</td>
<td>0.7</td>
<td>0.06</td>
</tr>
<tr>
<td>Jet fighter</td>
<td>0.4</td>
<td>0.07</td>
</tr>
<tr>
<td>Military cargo</td>
<td>1</td>
<td>0.08</td>
</tr>
<tr>
<td>Jet transport</td>
<td>1</td>
<td>0.09</td>
</tr>
</tbody>
</table>
To determine the tail arm, Raymer \cite{2} recommends using one of the three optimization methods. These methods recommend a value for the tail arm which produces the minimum (tail - tail cone) wetted area which means less drag and weight.

1. **Method 1: Initial tail sizing optimization considering the horizontal tail only:**
   The tail arm is calculated by using the following mathematical expression:
   \[
   l_{HT} = \sqrt{\frac{2 \cdot V_{HT} \cdot S_{Ref} \cdot C_{Ref}}{\pi (R_1 + R_2)}} \tag{81}
   \]
   where:
   - \(l_{HT}\) = horizontal tail arm in meter.
   - \(V_{HT}\) = horizontal tail volume.
   - \(R_1\) and \(R_2\) = the tail cone radius at the beginning and end respectively.
   - \(S_{Ref}\) = reference area (usually the wing area).
   - \(C_{Ref}\) = wing mean aerodynamic chord.

2. **Method 2: Initial tail sizing optimization considering the vertical tail only:**
   The tail arm is calculated using the following mathematical expression:
   \[
   l_{VT} = \sqrt{\frac{2 \cdot V_{VT} \cdot S_{Ref} \cdot b_{Ref}}{\pi (R_1 + R_2)}} \tag{82}
   \]
   where:
   - \(l_{VT}\) = vertical tail arm in meter.
   - \(V_{VT}\) = vertical tail volume.
   - \(b_{Ref}\) = wing span.

3. **Method 3: Initial tail sizing optimization considering the vertical and horizontal tail:**
   The tail arm is calculated using the following mathematical expression:
   \[
   l_T = \sqrt{\frac{2S_{Ref}(V_{VT} \cdot b_{Ref} + V_{HT} \cdot C_{Ref})}{\pi (R_1 + R_2)}} \tag{83}
   \]
   In the case, when the tail is not conventional tail, just the third method must be used.

After determining the tail arm and volume coefficient the tail area is calculated by using Equations \((84)\) and \((85)\).

\[
S_{HT} = \frac{V_{HT} \cdot S_{Ref} \cdot C_{Ref}}{l_{HT}} \tag{84}
\]

\[
S_{VT} = \frac{V_{VT} \cdot S_{Ref} \cdot b_{Ref}}{l_{VT}} \tag{85}
\]

The detailed information of the tail like MAC, tip chord, root chord, etc., are calculated based on the taper ratio, aspect ratio and sweep angle which are provided by the user and by using Equations from \((42)\) to \((48)\).
3.5.2 Aerodynamic Properties

1. Tail lift coefficient:
   According to Gudmundsson [2], the typical value for the horizontal tail aspect
   ratio ranges from three to five. This fact leads to that the lifting line theory
   which is used to determine the wing lift coefficient is not valid for the tail.
   Gudmundsson [2] presents three equations to estimate the lift curve slope. One
   of these equation is Equation 60 which is also presented by Nelson [30] and
   Bertin [14]. These equations were evaluated depending on the given data at
   the same reference. Equation (86) gives the best values, so it will be used to
   calculate the lift curve slope,

\[
C_{La} = \frac{2\pi \cdot AR}{2 + \sqrt{(\frac{AR\beta}{k})^2(1 + \frac{\tan^2 \angle C/2}{\beta^2}) + 4}}
\]  

where:
- \(\angle C/2\) = sweep-back angle at the mid-chord.
- \(\beta\) = Mach number parameter = \((1 - Mach^2)^{0.5}\)
- \(k\) = ratio of two dimension lift curve slope to \(2\pi\) = \(Cl\alpha \times (180/\pi)/(2\pi)\).

The lift coefficient at the zero angle of attack can be calculated by using the
following equation from Gudmundsson [2],

\[
C_{L0} = |\alpha_{ZL}| \cdot C_{La}
\]  

Where: \(\alpha_{ZL}\) is the zero lift angle of attack.

2. Tail pitching moment coefficient:
   To determine the tail pitching moment curve slope, Equation (77) is used.

3.5.3 Wetted Area

To calculate the tail wetted area and the inner volume, the presented method at
section (3.4.3) is used.

3.6 Fuselage

The main fuselage function is providing the required volume for the equipment and
the passenger. During the fuselage design, many other things must be considered as
the aircraft stability, minimize the drag.

According to the Gudmundsson [2], the fuselage shape can be a frustum-, tube-
or airship-hull. The fuselage shape will determine the used equation to estimate
the drag. The fuselage cross-section can be one of the three form, Figure (18).
These cross-sections are called from the left rectangular, elliptical and half-half cross
section. The used equations to calculate each cross section area and perimeter are
presented in appendix (A).
If the fuselage is divided into a number of segments, the segment shape will be one of the four presented type at Figure 19.

1. Shape A: the fuselage segment has the same cross section (area and type) in both sides. For instance, cylinder or rectangular cuboid,

2. Shape B: if the segment is a paraboloid.

3. Shape C: if the segment shape is cone or pyramid.

4. Shape D: When the fuselage segment has the same type of cross-section but different cross-section area at its sides. For instance, frustum or truncated cone.

The used equations to calculate the shapes side areas and volumes are presented in appendix (A).
3.7 Drag Model

Aircraft drag model is the mathematical description of the aircraft drag and how it changes at different condition. By using some simplification, the drag model can be written as in Equation (88), Gudmundsson [2].

\[ C_D = C_{D_0} + C_{D_f} + C_{D_i} + C_{D_w} + C_{D_{misc}} \]  

(88)

Where:

- \( C_D \) = total drag coefficient.
- \( C_{D_0} \) = pressure drag coefficient.
- \( C_{D_f} \) = skin friction drag coefficient.
- \( C_{D_i} \) = lift-induced drag coefficient.
- \( C_{D_w} \) = wave drag coefficient.
- \( C_{D_{misc}} \) = miscellaneous or additive drag coefficient.

The wave drag coefficient is ignored since the program is for general aviation aircraft only. The sum of \( C_{D_0}, C_{D_f} \) and \( C_{D_{misc}} \) is called minimum drag coefficient \( C_{D_{min}} \). Equation (88) can be written.

\[ C_D = C_{D_i} + C_{D_{min}} \]  

(89)

3.7.1 Lift-Induced Drag Coefficient

To estimate the lift-induced drag coefficient \( C_{D_i} \), there are two methods. The first one is the simplified method, Equation (90), which presented by Roskam [18], Anderson [21], Bertin [14], Torenbeek [3], Hale [20] and Gudmundsson [2]. The second one is the adjusted method, Equation (91), which according to Gudmundsson [2], Torenbeek [3] and Hale [20] gives better result. The main difference between them that the minimum value for drag coefficient is gotten when there is no lift in the simplified method. In the adjusted method, it is gotten at a certain value of the lift coefficient.

\[ C_{D_i} = \frac{C_L^2}{\pi \cdot AR \cdot e} \]  

(90)

\[ C_{D_i} = \frac{(C_L - C_{L_{minD}})^2}{\pi \cdot AR \cdot e} \]  

(91)

Where:

- \( C_{L_{minD}} \) = the lift coefficient where drag become a minimum.

\( C_{L_{minD}} \) value can be assumed to be equals to \( C_{l_{minD}} \) of the wing airfoil. Generally, it varies from about 0.00 to 0.30. Neither Gudmundsson [2], Torenbeek [3] nor Hale [20] gives an equation or method to determine \( C_{L_{minD}} \) value. Figure (20) presents the drag polar for an aircraft. It can be seen that the adjusted method gives more accurate results comparing to the simplified one.
3.7.2 Minimum Drag Coefficient

It is mentioned before that the sum of the pressure, skin-friction and miscellaneous drag coefficient is the minimum drag coefficient. Equation (92) and Equation (93) are the mathematical expression of these words. Each item in these two equations will be discussed.

\[
C_D = C_{D_0} + C_{D_f} = \frac{1}{S_{ref}} \sum_{i=1}^{N} C_{f_i} \cdot FF_i \cdot IF_i \cdot S_{wet_i}  
\]

\[
C_{D_{min}} = \frac{1}{S_{ref}} \sum_{i=1}^{N} C_{f_i} \cdot FF_i \cdot IF_i \cdot S_{wet_i} + C_{D_{misc}} 
\]

Where:

- \(S_{ref}\) = reference area.
- \(FF_i\) = form factor.
- \(IF_i\) = interference factor.
- \(S_{Wet}\) = wetted area.
- \(N\) = number of components.
- \(C_{f_i}\) = skin friction coefficient

The additive drag coefficient \((C_{D_{misc}})\) is the sum of a number of contribution sources. As examples of these sources are the trim drag, the landing gear drag, drag of canopy, cooling drag, etc. To calculate the additive drag coefficient many information must be known. Most of this information cannot be determined at this design point.
Skin Friction Coefficient

The skin friction coefficient $C_f$ depends on Reynold’s number, the surface type, flow type and the location of the transition point between the laminar and turbulent flow. For turbulent flow, the compressibility effects can be considered or ignored. Figure (21) shows the flow diagram for skin friction coefficient calculation for any component. Appendix (B) present all equations to calculate the skin friction coefficient. At those equations, Reynold’s number is the lowest between the normal number and the cutoff Reynolds number. Cutoff Reynolds number considers the surface quality while the normal Reynolds doesn’t. It can be calculated by using Equation (94), Gudmundsson [2], the equation was modified by the author.

$$R_{e-cutoff} = 38.21 \left( \frac{C \times 3.28084}{k} \right)^{1.053}$$  (94)

where:

- $C$ = reference length in meter.
- $k$ = skin roughness value. It can be determined from Table (23).

Figure 21: The flow diagram of the skin friction coefficient estimation method

Form Factor

Almost every one of the famous aircraft design books has it’s own equation to estimate the form factor ($FF$) value. Some of these equations consider the compressibility and sweep effects and some do not. The equations from Raymer, Shevell, Nicolai, Torenbeek and Jenkinson are presented in appendix (B). The fuselage form factor depends on the fineness ratio. Equation (95) is used to calculate the fuselage fineness ratio regardless of the fuselage shape.

$$f = l \sqrt[4]{A_{max} / \pi}$$  (95)

where:

- $l$ = fuselage length.
• $A_{max} =$ maximum fuselage cross section.

If the fuselage shape similar to the airship-hull shape, the fuselage form factor must be calculated by using the following equation, Gudmundsson [2].

$$FF = 3f + \frac{4.5}{\sqrt{f}} + \frac{21}{f^2}$$  \hspace{1cm} (96)

**Interference Factor**

When the aircraft components are gathered, they will constrain the airflow compared to the individual components. This will increase the local airspeed which increases the drag, Gudmundson [2]. Interference factor expresses this increasing. There are different methods to calculate the interface factor in all of Gudmundsson [2] Torenbeek [9] and Bertin [14]. The method from Gudmundsson [2] was chosen. The interference factor can be determined from Table (9).

<table>
<thead>
<tr>
<th>Component</th>
<th>IF</th>
</tr>
</thead>
<tbody>
<tr>
<td>High Wing</td>
<td>1</td>
</tr>
<tr>
<td>Mid Wing</td>
<td>1</td>
</tr>
<tr>
<td>Low Wing</td>
<td>1.1 - 1.4</td>
</tr>
<tr>
<td>Conventional Tail</td>
<td>1.04 - 1.05</td>
</tr>
<tr>
<td>Cruciform Tail</td>
<td>1.06</td>
</tr>
<tr>
<td>T-tail</td>
<td>1.04</td>
</tr>
<tr>
<td>V-tail</td>
<td>1.03</td>
</tr>
</tbody>
</table>
3.8 Component Weight Estimation

Different conceptual design books suggest different components weights estimation equations. Since those equations depend on the database from existing aircrafts, they might give a poor weights estimation when the advanced technology (composite structure) is used or other cases. This problem can be solved by using a method depends on the loads, not on the statistical equations.

At the used conceptual design tool at Linkoping University (Bex), a method from SAWE (Society of Allied Weight Engineers) paper 141 [22] was used with a method from Torenbeek book [24] to estimate the components weights as it is mentioned at Berry [23].

Torenbeek [3] states that his method can not be used if the material is not aluminum alloy. SAWE paper 141 [22] also presents another limitation. It is mentioned that if the wing structure type is not skin-multi web beam type the method must be modified. In addition to that, the weight penalty calculations depend on an old statistical data.

On the other hand, professor Patrick Berry [23] states that these methods give close value to the real values. The author believes that professor Berry modified these methods before using them. Since the only document of BeX (BeX manual [10]) does not explain how these methods were implemented, it was impossible to figure out the implementation way even after studying these methods from the original resources.

After some failed attempts to find a method to estimate the components weights which does not depend on the statistical equations, the decision was taken to choose one of the statistical estimation equations. All of Raymer [1], Gudmundsson [2], Torenbeek [3], Anderson [6] and Howe [19] present different statistical equations. Gudmundsson presents both Raymer’s and Nicolai’s equations. He recommends evaluating these equations against the existing aircraft from the same category of the designed aircraft to find correction factors.

The chosen method is Raymer’s equations from [1] and [2] for the following four reasons:

- He suggests fudge factors for his equations. Table [10] shows these factors from Raymer [1].

- Raymer equations give the lowest predicted fuselage and the wing weights according to SAWE paper 3597 [25]. The difference between the predicted weights of different methods is small. The problem with the statical equations is that they give higher predicted values than the real, since the material are being improved and the structure weight is going down. Therefore, the method which gives lower values is better (less error).

- Some equations in the other methods require information and variables which can not be determined before this step.

- Anderson’s equations are so simple and depend only on the wetted area of the component. The equations do not consider many affected variables. For instance, the equations do not consider the fuel tank position.
The suggested fudge factors by Raymer [1] can be used or the user can find his own correction factors. These equations were evaluated by the author for ultralight aircraft. The results show that the correction factor is around 0.78 (for the total aircraft weight) which is lower than Raymer’s suggestion. The chosen equations can found in appendix (C). Gudmundsson [2] presents physical method. It is called "Direct Weight Estimation Method". This method is not complete and can not be used since it needs correction factors and also when this method was used for the fuselage weight estimation, it gave an error result. Dr.Gudmundsson was asked by the author about this method and how to extend it to the fuselage. He answered that this method requires so many other components weights to be calculated before using it. Also, the number of spars, ribs, and stringers must be determined. These detail information can not be determined in this step, therefore this method is ignored. In his answer, he recommends using the statistical equations with correction factors.

<table>
<thead>
<tr>
<th>Category</th>
<th>Weight group</th>
<th>Fudge factor</th>
</tr>
</thead>
<tbody>
<tr>
<td>Advanced composites</td>
<td>Wing</td>
<td>0.85 - 0.90</td>
</tr>
<tr>
<td></td>
<td>Tails</td>
<td>0.83 - 0.88</td>
</tr>
<tr>
<td></td>
<td>Fuselage/nacelle</td>
<td>0.90 - 0.95</td>
</tr>
<tr>
<td></td>
<td>Landing gear</td>
<td>0.95 - 1.00</td>
</tr>
<tr>
<td>Braced Wing</td>
<td>Wing</td>
<td>0.82</td>
</tr>
<tr>
<td>Wood Fuselage</td>
<td>Fuselage</td>
<td>1.6</td>
</tr>
<tr>
<td>Steel tube fuselage</td>
<td>Fuselage</td>
<td>1.8</td>
</tr>
</tbody>
</table>
4 Implementation Method And Results

The knowledge from the theory part was implemented in Microsoft Excel. The implementation strategy consists of the following items.

1. Creating functions by using the Excel VBA (Visual Basic for Applications), when the equations are used frequently. For example: the atmospheric properties.

2. Creating functions or macros if the used method is complex or need an iterative process to be solved. For example: the lifting line theory.

3. Creating macros in VBA to control the Excel sheets and to do some calculations. For example: the propulsion macro.

4. Using colors to distinguish between the cell types. Yellow means that the value is calculated in the active sheet. Grey means that the value comes from a different sheet. While white means that the cell is an input cell.

5. Giving the user the possibility to use the estimated value or to insert his values in some sections. As well as the possibility to select the used method or equation. For example: components weight and the drag calculations equations respectively.

6. Reducing the number of inputs as much as possible. For instance, by the use of drop-down lists, whenever the input cell is a value from a list.

7. Covering different configurations whenever possible.

8. Extend use of comments to explain the cells meaning and values when applicable. Writing notes at the top of sheets to highlight important information.

9. Creating on-line links to give the user information. When clicking a link cell, a PDF file will be opened. Those PDF files were created by the author. They contain some explanations or figures to help the user to understand or to estimate the topic.

10. If the used values in the sheet come from different sheets, this will be mentioned. For instance, the cruise altitude is determined in the "Main-Information" sheet and the input-cell name is "Cruise-Altitude". To use the cruise altitude in the "Propulsion" sheet, a cell in "Propulsion" sheet is named "Cruise-Altitude-Propulsion" and is connected to the original cell. This will allow the user to know all the used variables inside any sheet and can be very useful for future development of the program.

11. Well documented code: writing comments in the macros and in the functions to explain the code and to make any future development easier.
12. Giving names for the cells. The naming strategy is to give the cell the name of the variable and at the end adding the sheet name. For instance, "Take-off-Altitude-Sizing" for the takeoff altitude in the "Sizing" sheet.

The data flow inside the program is presented in Figure 22. The program starts with the "Main-information" sheet which contains general information and some performance requirements. The initial weight estimation phase can be used or ignored depending on the users desire. This phase is only used one time to give the first weight estimation. Then, the data and informations goes to the sizing phase which consists of tow parts, the carpet plot and the constrain diagram. This phase is used mainly to determine the wing loading and required thrust. Then the required thrust value is exported to the propulsion phase, while the wing loading goes to the geometry phase where the aircraft shape and dimensions are defined. Two sheets are used to define the geometry. After completing both (propulsion and geometry) the flow completes to the drag estimation and components weight estimation phases.

![Figure 22: The data flow diagram in the program](image)

This report doesn’t include specific results for a specific study case. It shows the ways of presenting the result inside the program. Therefore, both the implementing methods and the results are combined in one section in the report. Each phase from the flow diagram is discussed in detail in a separate subsection. Except the geometry phase which is divided into three part (wing, tail and fuselage) and every part is discussed and considered as a separate subsection. The used methods are explained first, then the how the results presentd in the program are demonstrated. Almost all of the results are presented by using cells inside the program, therefore they were not included inside the report. While, The used table and figures to preset the results were included. At the end of each subsection the used VBA scripts, their locations and their functions are presented. If there are different options for the figures or the tables depending on the input values, just one example is presented. For instance, the used figure to present the available thrust at different altitudes form the jet engines will change depending on the engine type (turbojet, turbofan or turboprop). just one example is included in the report. There is no reason to show the all figures for different jet engine types. It is worth to mention that after implementing any method the results were evaluated.
against one of the following two options to make sure that no wrongs or mistakes are done during the method application:

1. The numerical examples in the references, if they are given.
2. The real values for existing aircraft.

If the references don’t give numerical examples, and the real values for existing aircraft are unknown, the author makes sure that the method gives realistic values. A good example of this, Oswald’s span efficiency estimation methods.
4.1 Main Information

The first excel sheet in the program is "main-Information" sheet. This sheet is used to defined the general data and values. It is divided into three sections, general information, crew information and airport information. The desired stall speed, the required cruise condition (altitude and velocity) and the service ceiling altitude must be determined by the user in the main information section. This section also includes two input-cells to determine the engine type and the expected gross weight.

The crew information section includes three input cells, the number of crew, the number of passenger and the person mass in (kg). The federal regulation (FAR-23) assumes that the average weight of a person is (170)lb around (77.08) kg. Gudmundsson [2] states that this weight is too light since the average person weight increased in the last years. Raymer [1] assumes the average weight of person is (85) kg. For this reason, the author decided to make this value an input value.

The airport information section includes just two inputs, the airport altitude and the airport ground type. The last one is used to estimate the ground friction coefficient. Table (11) presents the ground friction coefficients for different ground types. Table (12) presents the used VBA codes in "Main-Information" sheet, their functions and their locations.

Table 11: Ground friction coefficient from Sadraey [7] with some modifications

<table>
<thead>
<tr>
<th>Ground Type</th>
<th>Friction coefficient</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dry asphalt</td>
<td>0.04</td>
</tr>
<tr>
<td>Wet asphalt</td>
<td>0.05</td>
</tr>
<tr>
<td>Icy asphalt</td>
<td>0.02</td>
</tr>
<tr>
<td>Turf</td>
<td>0.06</td>
</tr>
<tr>
<td>Grass</td>
<td>0.85</td>
</tr>
<tr>
<td>Soft ground</td>
<td>0.03</td>
</tr>
</tbody>
</table>

Table 12: The used VBA codes in the "Main-Information" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmospher-Function&quot; Module</td>
<td>Calculate the atmospheric properties</td>
</tr>
<tr>
<td>Sheet script</td>
<td>-</td>
<td>&quot;Main-Information&quot; sheet</td>
<td>Estimate the ground friction coefficient</td>
</tr>
</tbody>
</table>
4.2 Initial Take off Weight Estimation

The equations from section 3.1 were implemented in two separated excel sheet. The first one is "Weight-Estimation" sheet. This sheet can be used to estimate the takeoff weight for non electric aircraft. The second sheet is "Weight-Estimation-Electric" sheet, for electric aircraft. The mentioned sheets are not connected to any other sheets. This in order to give the user the possibility to use or to avoid this sheets, especially in the iterative progress.

Each one of the above mentioned sheets is divided into four sections. The first one is the main information section. In this section the user should provide the main information of the aircraft like the crew and load weights, the aircraft category and the engine type, if the aircraft is not an electric aircraft.

The second section is the fuel fraction section (Battery fraction section). It begins with estimating the maximum lift to drag ratio. Then, the fuel fraction is estimated depending on the mission profile which can be simple or commercial. The commercial mission refers to the mission which has two loiter sections and two cruise section. If the mission profile is simple, the range in "Cruise2" and the endurance in "Loiter2" must be equal to zero.

The third section is the empty weight fraction section. This section estimates the constants in Equation (3) automatically depending on the aircraft type. This section is controlled by a code which is written in the sheet script.

The fourth and the last section is the plot section. Since the empty weight fraction is a function of the total weight, an iterative progress must be used to solve either Equation (1) or Equation (2). The user has to provide the minimum guessed weight and the weight step. The program will generate Figure (23), the horizontal axis present the number of iteration.

![Figure 23: Initial weight estimation plot from the program](image)

The intersection between the two lines is the initial weight. An on-line link was added to the "Weight-Estimation" sheet, to help the user to estimate the fuel consumption in both loiter and the cruise segments. The included information in the on-line PDF file is Tables 3.3 and 3.4 in the Raymer [1].
Table (13) shows the used VBA scripts in this section with their functions and locations.

Table 13: The used VBA script in the initial weight estimation

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sheet script</td>
<td>&quot;Weight_Estimation&quot;</td>
<td>-</td>
<td>Determine the used constants values in the empty weight fraction equation</td>
</tr>
<tr>
<td>Sheet script</td>
<td>&quot;Weight_Estimation_Electric&quot;</td>
<td>-</td>
<td>Determine the used constants values in the empty weight fraction equation</td>
</tr>
</tbody>
</table>
4.3 Initial Sizing

According to the section (3.2), the constraint diagram and the stall-cruise speeds carpet plot are used to determine the required thrust and the wing loading.

4.3.1 Constraint diagram

Constraint diagram is located at the "Sizing" sheet in the program. The input data of this sheet comes from two sources, "Main-Information" sheet and the user. The sheet includes two links to help the user to estimate important values. The first link shows some typical aerodynamic characteristics of the selected class of the aircraft. The link includes Table 3-1 at Gudmundsson [2]. This link can be used to estimate the minimum drag coefficient $C_{D_{\text{min}}}$, the lift coefficient during take-off run $C_{L_{\text{TO}}}$ and the drag coefficient during the take-off run $C_{D_{\text{TO}}}$. The second link presents the climb performance characteristics of different aircraft. The link includes Table 18-1 at Gudmundsson [2]. This link gives the user more information and better understanding of the rate of climb and vertical speed during the climb.

The user can choose the wing loading range and the number of points in this range as it is shown in Figure (24). The user must be careful during the wing loading range and the step selection, that the number of points ((Range End - Range start)/Step) must be an integer number or the program will not work.

![Figure 24: The wing loading range and step from the program](image)

The constraint equations from Equation [2] to Equation [13] were converted to functions. These functions are called inside the "Sizing" macro which is used to control almost the whole "Sizing" sheet. By running the macro, the old rows, values and the sizing diagram’s series will be deleted. Then, new rows will be generated and the functions will be called to calculate the new thrust (power) to weight ratios. Figure (25) shows an example of a constraint table inside the program which is used to present the calculated values in the macro. Figure (25) also includes the normalized thrust (power) values (to the sea level), which are calculated according to the equations at section 2.3.1 with some modification.

![Figure 25: An example of constraints tables from the program](image)
After completing the constraint tables generating, the program calculates the required maximum lift coefficient to achieve the desired stall speed and other three regulation limits. These limits are 30 (m/s) for LSA (light sport aircraft) regulation, 23 (m/s) for FAR (Federal Acquisition Regulation) and 18.05 (m/s) for the ultralight aircraft.

Then the program generates the constraint diagram, Figure (26). The wing loading (X axis) unit is Newton per meter square while the vertical axis unit is Wat/Newton or Newton/Newton depending on the engine type. The vertical line (the black line) presents the selected wing loading value by the user. This value is used to calculate the required landing distance.

![Constraint Diagram](image)

**Figure 26: The constraint diagram from the program**

The "Sizing" sheet also calculates the following parameters:

- The lift induced drag constant by using Equation (97). This value is used in the constraints equations.
  \[ k = \frac{1}{\pi \cdot AR \cdot e} \]  
  \( (97) \)

- The required lift coefficient during cruise by using Equation (98). This value will be used later in the wing design.
  \[ CL_{req} = \frac{2 \cdot W}{\rho \cdot S_{ref} \cdot V^2} \]  
  \( (98) \)

- The available thrust or power at cruise condition by using the equations from section 2.3.1. This value is used to generate the carpet plot.

The program gives the user the possibility to modify the constraint diagram axis directly form the sizing sheet. Table [14] present the used VBA script in this section with their functions and locations.
### Table 14: The used VBA script in the "Sizing" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmosphere-Function&quot; Module</td>
<td>Calculate the atmospheric properties</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Sizing&quot;</td>
<td>&quot;Sizing&quot; Module</td>
<td>Modify the rows. Create the constraint tables. Create the constraint diagram.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Sizing_ADD_Color&quot;</td>
<td>&quot;Sizing&quot; Module</td>
<td>Color the constraint diagram. Add the vertical line</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;rate_off_climb&quot;</td>
<td>&quot;Function_rateoff_climb&quot; Module</td>
<td>-</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;ServiceCeiling&quot;</td>
<td>&quot;Function_Service_Ceiling&quot; Module</td>
<td>-</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;TakeOffDistance&quot;</td>
<td>&quot;Function_takeoff_distance&quot; Module</td>
<td>-</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;velocity_turn&quot;</td>
<td>&quot;Function_velocity_turn_sizing&quot; Module</td>
<td>-</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;cruiseairspeed&quot;</td>
<td>&quot;Function_cruise_airspeed&quot; Module</td>
<td>-</td>
</tr>
<tr>
<td>Sheet script</td>
<td>-</td>
<td>&quot;Sizing&quot; Module</td>
<td>Modify the constraint diagram axes limits</td>
</tr>
</tbody>
</table>

#### 4.3.2 Cruise-Stall Speeds Carpet Plot

The cruise-stall speeds carpet plot is located at "Carpet-plot" sheet in the program. As it was mentioned in the section (3.2.2), this plot is used to help the user to select the correct wing loading value from the constraint diagram. "Carpet-plot" sheet and "Sizing" sheet are highly related. This fact reduces the needed inputs in "Carpet-plot" sheet to just, the maximum lift coefficient range and the wing area range.

The first step in the carpet plot generating is determining the stall speed at different values for the wing area at different maximum lift coefficients. It is done inside the program by using Equation (17) and the results are demonstrated by using a table. Figure (27) shows the used table inside the program.

![Figure 27: The stall speed table in "Carpet-Plot" sheet](image-url)
The second step is calculating the maximum cruise speed. This calculation depends on the propulsion system type.

- **Propeller driven aircraft:**
  If the aircraft is a propeller-driven aircraft, the maximum cruise velocity can be calculated by using Equation [18]. To solve this equation, an iterative process must be implemented. A function was built to do this mission. The main function body is presented in section (19.2.14) in Gudmundsson [2]. The author modified this function to be able to use it in the program. It is highly recommended to read sections (19.2.10), section (19.2.14) and the example (19-11) in the same reference.

- **Jet aircraft**
  Equation [19] is used to calculate the maximum cruise velocity.

A macro was built inside the program to choose the correct equation between Equation [18] and Equation [19] depending on the engine type. The results are presented by using a table. Figure (28) shows that table.

<table>
<thead>
<tr>
<th>Reference Area (S)</th>
<th>np</th>
<th>Vmax</th>
</tr>
</thead>
<tbody>
<tr>
<td>5</td>
<td>0.8</td>
<td>90.24289355</td>
</tr>
<tr>
<td>10</td>
<td>0.8</td>
<td>73.03458521</td>
</tr>
<tr>
<td>15</td>
<td>0.8</td>
<td>647097437</td>
</tr>
<tr>
<td>20</td>
<td>0.8</td>
<td>58.62860986</td>
</tr>
<tr>
<td>25</td>
<td>0.8</td>
<td>54.56629297</td>
</tr>
<tr>
<td>30</td>
<td>0.8</td>
<td>51.44214746</td>
</tr>
<tr>
<td>35</td>
<td>0.8</td>
<td>48.93233374</td>
</tr>
</tbody>
</table>

**Figure 28:** The Maximum cruise speed table in "Carpet-plot" sheet

By finishing this step, all the needed values to generate the carpet plot are determined. Next step is preparing these values for plotting. The preparation is done according to the presented instructions at Gudmundsson [2]. Figure (29) presents the carpet plot shape in the program.

**Figure 29:** The carpet plot in the program
Table 15 presents the used VBA scripts in "Carpet-Plot" sheet.

Table 15: The used VBA script in the "Carpet-Plot" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmospher-Fuction&quot; Module</td>
<td>Calculate the atmospheric properites</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Prop-Vmax&quot;</td>
<td>&quot;Function-Prop-Vmax&quot; Module</td>
<td>Calculate the maximum cruise velocity for the propeller aircrafts</td>
</tr>
</tbody>
</table>


4.4 Propulsion System

The "Propulsion" sheet arrangement in the program is modified and changed depending on the aircraft type. A macro with name "Propulsion" is responsible of controlling and modifying this sheet. Mainly, there is two configurations of the sheet arrangement, propeller-driven aircraft configuration, and jet aircraft configuration. At the beginning, a function to calculate the available power or thrust depending on the engine type was built and its name is "Engine-thrust". This function includes the presented Equations at the section (3.3.1) in this report.

4.4.1 Propeller Driven Aircraft Configuration

This arrangement is used if the engine is an electric or a piston engine. The "propulsion" sheet in this arrangement is divided into five subsection.

1. The main informations:

   This subsection includes the general propulsion system information, like the engine type, the required power and the number of engines. The user must determine if the engine is known or not. If the engine is not known the program will calculate the power per engine and the engine weight. In this case (the engine is not known) the power per engine is equal to the required power divided by the number of engines.

   In general, this subsection is the same for both configurations with one difference. If the configuration is propeller-driven aircraft, this subsection will include an input cell to determine the engine revolutions per minute (RPM). Else if the configuration is jet aircraft, this subsection will include an input cell to determine if the result must be shown against Mach number or the velocity.

2. The power changing with altitude:

   This subsection appears only when the used engine is a piston engine. To determine the available power at different altitudes, the function "Engine-Thrust" is called inside the "Propulsion" macro. The results are presented by using a table and a figure. Figure (30) shows an example of how the results are plotted in the program.

![Available Power / Altitude](image)

Figure 30: The available power changing with altitude
3. Propeller anatomy and available thrust:

As it was mentioned in the theory part, this program is not a propeller design program. The program concentrates on helping the user in propeller choosing and determines the propeller efficiency at different flight conditions. A macro with name "Propeller" is used to control this subsection.

This subsection (propeller anatomy) starts by defining some input values such as the propeller material, propeller kind, number of blades and if the propeller is known or not. By running the macro "Propeller", the section will be adjusted depending on the input values. If the propeller is known, the propeller geometry information and properties \((C_p, C_T, \eta_P)\) equations must be inserted by the user. Figure (31) shows the propeller data section in this case. The constants \(a, b\) and \(c\) in the figure are the constants in the propeller properties equations like the equations (36), (37) and (38). Equation (99) gives an example for more clarification. Actually, in the further calculations, just the propeller efficiency equation and its diameter are needed.

\[
\text{Property} = a + b \cdot J + c \cdot J^2
\]  

(99)

Figure 31: Propeller Data section in the program if the propeller is known

On the other hand, if the propeller is unknown, the program will estimate the propeller diameter, required pitch distance and the pitch angle. These calculation are done inside the "Propeller" macro by using the equations from Equation (30) to Equation (35) except Equation (33).

To determine the available thrust, the propeller efficiency must be determined before. The efficiency equation is used directly, if the propeller is known. If the propeller is unknown the program assumes that the propeller is a constant speed propeller. To estimate the efficiency in this case, the momentum theorem is used. A function with an iterative process was built for this theorem.

By running the macro "Propeller", it calculates the propeller efficiency and the available thrust at twelve different velocities at both sea and cruise level. The results are presented by using a table, See Figure (32). The macro also plots the available thrust at both cruise and sea level against the velocity. Figure (33) shows an example of the available thrust against the velocity at cruise condition.

4. Certain condition:

In this subsection, the propeller efficiency and the available thrust at certain flight conditions are calculated. In the previous subsection, the temperature off-set from the standard atmosphere is ignored, while in this section it is considered. A special macro with name "Special-Case-Propeller" controls this
subsection. The calculations are done as it was explained in previous subsections but just for one point.

5. Engine weight and nacelle dimensions:

The user must determine the nacelle dimensions. The input-cells for the nacelle are the height, the width, the length and the cross section shape. Figure (34) shows the meaning of the first three inputs. The program assumes that the cross section is either elliptical or rectangular and uses the inserted values to calculate the nacelle’s wetted area, the fineness ratio and the cross section area.

The engine weight must be provided by the user except if the engine is unknown and it is a piston or a turbofan engine. In this case, the Equation (40) or Equation (41) are used depending on the engine type to estimate the engine weight.

4.4.2 Jet Aircraft Configuration

This arrangement is used if the engine is a jet engine. The "propulsion" sheet in this arrangement is divided into four subsection.

1. The main informations

This section was described before.

2. Thrust changing with altitude and velocity

<table>
<thead>
<tr>
<th>Velocity</th>
<th>Efficiency</th>
<th>Available Thrust (*)</th>
<th>Available Thrust</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>Infinite</td>
<td>Infinite</td>
<td>Infinite</td>
</tr>
<tr>
<td>6.5</td>
<td>0.23</td>
<td>1750.36</td>
<td>1750.36</td>
</tr>
<tr>
<td>13</td>
<td>0.35</td>
<td>1334.66</td>
<td>1334.66</td>
</tr>
<tr>
<td>19.5</td>
<td>0.46</td>
<td>1183.25</td>
<td>1183.25</td>
</tr>
<tr>
<td>26</td>
<td>0.53</td>
<td>1059.91</td>
<td>1059.91</td>
</tr>
<tr>
<td>32.5</td>
<td>0.64</td>
<td>917.00</td>
<td>917.00</td>
</tr>
<tr>
<td>39</td>
<td>0.71</td>
<td>853.31</td>
<td>853.31</td>
</tr>
<tr>
<td>45.5</td>
<td>0.76</td>
<td>826.59</td>
<td>826.59</td>
</tr>
<tr>
<td>52</td>
<td>0.81</td>
<td>704.24</td>
<td>704.24</td>
</tr>
<tr>
<td>58.5</td>
<td>0.84</td>
<td>647.39</td>
<td>647.39</td>
</tr>
<tr>
<td>65</td>
<td>0.86</td>
<td>591.40</td>
<td>591.40</td>
</tr>
</tbody>
</table>

Figure 32: The used table to present the propeller efficiency and the available thrust

Figure 33: The available thrust plot against the velocity at cruise condition
The available thrust from engine depends on both altitude and the velocity (Mach number) as it was presented in section (3.3.1). The maximum available thrust is determined at twelve different points at both sea and cruise level. These calculations are done inside "Propulsion" macro by calling "Engine-Thrust" function. The results are presented by using one table and one figure for both sea and cruise levels. Figure (35) shows an example for the available thrust plot inside the program.

![Figure 35: The available thrust plot for turbojet engine at cruise and sea levels](image)

3. Certain condition

In this subsection the available thrust from the engine at certain condition is calculated. At the calculation in this subsection, the temperature offset is taken into account. A macro with name "Special-Case-Jet" controls this subsection.

4. Engine weight and nacelle dimensions

This section was described before.

Table (16) present the used VBA script in the "Propulsion" sheet.
Table 16: The used VBA script at "Propulsion" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmosphere-Fuction&quot; Module</td>
<td>Calculate the atmospheric properties</td>
</tr>
</tbody>
</table>
| Macro       | "Propulsion"        | "Propulsion" Module | 1. Modify the propulsion sheet by hiding and showing the sections depending on the engine type.  
|             |                     |                   | 2. Calculate the available power (thrust).  
|             |                     |                   | 3. Generate the thrust and power plots.  
|             |                     |                   | 4. Delete old values and figures to protect the program and prevent the errors |
| Macro       | "Propeller"         | "Propeller-macro" Module | 1. Modify the propeller section depending on the propeller type.  
|             |                     |                   | 2. Calculate the propeller efficiency  
|             |                     |                   | 3. Calculate the available thrust  
|             |                     |                   | 4. Generate the available thrust plots. |
| Function    | "Engine-Thrust"     | "Propulsion-Function" Module | Calculate the available thrust for different kinds of engines at different flight conditions.  
|             |                     |                   | Calculate the available power if the engine is piston engine. |
| Function    | "Prop-Efficinecy"   | "Propulsion-Function" Module | Calculate the constant speed propeller efficiency. |
| Macro       | "Special-Case-Propeller" | "Propulsion" Module | Calculate the propeller efficiency and the available thrust at certain condition for the propeller aircraft. |
| Macro       | "Special-Case-Jet"  | "Propulsion" Module | Calculate the available thrust at certain condition for the jet engines. |
| Sheet script| -                   | "Propulsion" Sheet | 1. Protect the program to be used in a wrong way.  
|             |                     |                   | 2. Calculate the nacelle’s wetted area, fineness ratio and cross section.  
|             |                     |                   | 3. Calculate the engine weight. |

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4.5 Wing

A separate sheet is used to design and study the wing. This sheet is divided into four subsections in the same way section (3.4) is divided. At the end of this section, there is a subsection to explain the exporting wing data to "Geometry" sheet.

4.5.1 Wing layout and equivalent wing

This subsection has two main functions. The first one is defining the wing geometry layout inside the program. The second mission is generating the equivalent trapezoidal wing if the original wing is not trapezoidal. This subsection begins with the main information like the wing area (comes from "Sizing" sheet), the wing type and the wing shape.

Wing geometry defining way inside the program changes according to the wing shape. The program covers three wing shapes.

1. Trapezoidal wing

To defined this wing, four variables are required. Three of them are mandatory which are the wing area, the taper ratio and the leading edge sweep angle. The fourth variable can be either the wing span or the wing aspect ratio. the program gives the user the possibility to choose the desired way to defined this variable.

After determining the previous variables, the program will plot the wing and calculate its detailed information. this done by running the “Trapezoidal-Wing” macro. The calculated detailed information is presented in a table. Figure [36] and Figure [37] show the wing plot and the detailed information table respectively.

![Figure 36: An example of the trapezoidal wing figure in the program](image1)

![Figure 37: Trapezoidal wing detailed information table in the program](image2)

2. Compound wing

To define this kind of wing, the first step is to define the number of segments which is unlimited. By running "Compound-Wing" macro, the program generates a table to determine each segment variables. These variables are the
root chord, the tip chord, the span and the sweep angle. Figure (38) shows an example of the generated table to determine the segments variables.

<table>
<thead>
<tr>
<th>Segment</th>
<th>Root chord</th>
<th>Tip chord</th>
<th>span</th>
<th>sweep angle (LE)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>3</td>
<td>2</td>
<td>2.5</td>
<td>15</td>
</tr>
<tr>
<td>2</td>
<td>2</td>
<td>1.5</td>
<td>5</td>
<td>5</td>
</tr>
<tr>
<td>3</td>
<td>1.5</td>
<td>1.3</td>
<td>2</td>
<td>15</td>
</tr>
</tbody>
</table>

Figure 38: The table of the Compound wing segments

In the next step, two macros are used. The first one is "Compound-Chart-Wing" macro which is used to plot the compound wing, as in Figure (39). The second macro is "Equivalent-Wing" macro which is used to generate the equivalent trapezoidal wing, plot the equivalent wing and to calculate the detailed equivalent wing information. The equivalent wing will plot in the same figure which includes the original compound wing plot, as in Figure (40). A table as in Figure (37) is used to present the equivalent trapezoidal wing detailed information.

Figure 39: An example of the compound wing figure in the program

Figure 40: An example of the compound wing and its equivalent wing figure.

3. Semi elliptical wing

To define this kind of wing, three variables must be determined, the root chord length, the half span length and the leading edge. The program gives two options to define the wing leading edge, manually or by using an equation. In any case, The user must determine the number of used points to define the leading edge. If the user chooses the manual way, the program will generate a three column table with initial values to give the leading edge a curvy shape.
and to ease the modification. See Figure (41). The values in the table depend on the root chord and span values.

<table>
<thead>
<tr>
<th>X (Right side)</th>
<th>Y</th>
<th>X (Lift side)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.00</td>
<td>0.00</td>
<td>0.00</td>
</tr>
<tr>
<td>2.50</td>
<td>0.31</td>
<td>-2.50</td>
</tr>
<tr>
<td>5.00</td>
<td>1.25</td>
<td>-5.00</td>
</tr>
<tr>
<td>7.50</td>
<td>2.81</td>
<td>-7.50</td>
</tr>
<tr>
<td>10.00</td>
<td>5.00</td>
<td>-10.00</td>
</tr>
</tbody>
</table>

Figure 41: The generated table to define the leading edge in the manual way

On the other hand, if the choice is to use an equation to define the leading edge, a smaller table with four input cells will appear, Figure (42). These cells are the leading edge equation constants, as the Equation (100) shows.

\[ y = a \cdot x^3 + b \cdot x^2 + c \cdot x + d \]  \hspace{1cm} (100)

After defining the wing geometrical information, two macros are used to plot the wing and to generate the equivalent trapezoidal wing. In a similar way to how the compound wing is treated (the used macros names are "Semi-Elliptical-Wing" and "Equivalent-Wing"). Figure (43) shows an example for an semi elliptical wing plot and an example for elliptical wing with its equivalent trapezoidal wing plots.

Figure 43: A semi elliptical wing plot (up) and the wing with its equivalent wing (down)
Table (17) shows the used VBA scripts in the wing layout and equivalent wing section.

**Table 17: The used VBA scripts in the wing layout and equivalent wing subsection**

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
</table>
| Macro| "Wing-Anatomy"        | "Wing-Anatomy" Module| 1. Modify the sheet depending on the Wing shape  
                          2. Delete old figures and values to prevent the duplications or errors |
| Macro| "Trapezoidal-Wing"    | "Wing-Anatomy" Module| 1. Generate the trapezoidal wing plot  
                          2. Calculate the detail information  
                          3. Save the plotting information in the "Matrix-Solve" sheet |
| Macro| "Compound-Wing"       | "Compound-Wing" Module| Generate the segments table                                                |
| Macro| "Compound-Wing"       | "Compound-Chart-Wing" Module| 1. Create the compound wing plot  
                          2. Save the plotting information in the "Matrix-Solve" sheet |
| Macro| "Semi-determine-way"  | "Wing-Anatomy" Module| 1. Control the semi elliptical section  
                          2. Create the leading edge table if the determine way is manual |
| Macro| "Semi-Elliptical-Wing"| "Wing-Anatomy" Module| 1. Create the semi elliptical wing plot  
                          2. Save the plotting information in the "Matrix-Solve" sheet |
| Macro| "Equivalent-Wing"     | "Equivalent-Wing" Module| 1. Calculate the equivalent trapezoidal wing  
                          2. Create the equivalent wing plot |
| Macro| "Equivalent-Wing-equation"| "Equivalent-Wing-Module"| This macro is called inside the "Equivalent-Wing" macro and it is used to calculate the equivalent trapezoidal wing for the semi elliptical wing when the leading edge is determined by using the equation option. |

It can be noticed in the table that some macros’ functions are to save the plotting information in the "Matrix-Solve" sheet. This information is used to export the wing shape to the "Geometry" sheet. The exporting process is described in detail in section (4.5.5).
4.5.2 Aerodynamic Properties

This sections includes the wing aerodynamic properties. It is divided into four subsection.

1. Airfoil Data

The user has to determine the used airfoil data in this subsection. It can be noticed that there are two input cells for the maximum lift coefficient. This is because the maximum lift coefficient depends on the Reynolds number which depends on the reference length. See Equation (101). The reference length in the tip is not the same as in the root. To calculate the average maximum lift coefficient for the airfoil, Equation (102) from Gudmundsson [2] is used.

\[
Re = \frac{\rho \cdot V \cdot C}{\mu} \tag{101}
\]

\[
(Cl_{\text{max}})_{\text{ave}} = (C_{l_{\text{max}}})_{\text{root}} + \frac{2MAC}{b} [(C_{l_{\text{max}}})_{\text{tip}} - (C_{l_{\text{max}}})_{\text{root}}] \tag{102}
\]

where:

- \( V \) = Air velocity.
- \( \rho \) and \( \mu \) = Air density and viscosity respectively.
- \( C \) = reference length.
- \( MAC \) and \( b \) = Wing mean aerodynamic chord and span respectively.

2. Maximum Lift Coefficient

As it was mentioned in the theory part, Two methods were chosen to estimate the maximum lift coefficient, Rapid Estimation method and USAD DATCOM method. A macro was built to control this subsection. The program provides two links to estimate the maximum lift ratio and the Mach correction factor. These links include Figure (13) and Figure (14).

3. Wing Coefficients

Implementing the lifting line theory required a macro to be built. As it was mentioned in the theory part, Equation (61) is the heart of this theory. This equation is used to generate a system of equations as in Figure (44).

Solving this system requires using matrix inverse process. VBA doesn’t support this function. Due to the time lack, the author decided to use an excel sheet in the background to inverse the matrix.
The lifting line theory calculates the lift coefficient value $CL_1$ during the cruise. This value must be equal to or bigger than the required lift coefficient at cruise $CL_{req}$ which is calculated in "Sizing" sheet. Changing the wing setting angle or the wing twisting angle can be used to get the required lift coefficient. To determine the lift curve slope, each segment’s angle of attack is increased by one degree inside the macro and then the lifting line theory is used to calculate the new lift coefficient value $CL_2$. After completing these calculations, Equations (67), (65) and (63) are used to determine the angle of attack at cruise, the lift curve slope and the lift coefficient at zero angle of attack respectively. This subsection includes also the moment curve information and the stall speed. Equations (77) and (75) are used to do these calculations respectively.

4. Oswald Span Efficiency Factor.

The program gives the user the ability to choose one of the presented methods in the theory or to take the average value of all of them. A macro is used to control this section to hide and show the methods. One link was added to the program to help the user to determine the lift-induced drag factor $\delta$. The link includes Figure (9).

4.5.3 Wetted Area and Volume

The user can choose between either to calculate the wetted area with correction or not. If the choice was with correction, the user must insert the values of the new root chord and the fuselage average width at the connection point. One link was added to the sheet to explain the new root meaning. The link includes Figure (15).

4.5.4 Lift Distribution Over the Wing

Implementing the wing lifting line theory allows calculating the lift coefficient distribution over the wing. The results are presented in a table and a figure. The table number of rows is adjusted depending on the number of segments. The user has the option to choose between 10, 15 or 20 segments for each half span. Figure (45) shows how the results are plotted in the program. Sadraey [7] states that the lift coefficient distribution over the half span must be elliptical. The major affected parameter on the lift coefficient distribution shape is the taper ratio.

![Figure 45: The lift coefficient distribution over the half span](image-url)
4.5.5 Export The Wing Geometry Data

The main reason behind using a separate excel sheet to design the wing is to simplify the program understanding and to give the user a better view on the wing design process. But at the same time, it has a drawback point. The user can be confused and some unnecessary complexity can occur since the aircraft geometry informations are defined in two excel sheet, not one. To overcome this negative point, the wing geometry information are exported from "Wing-Anatomy" sheet to "Geometry" sheet. This process is done by using a macro with the name "Wing-Export". This means that the user will design the wing on the "Wing" sheet and then by running the macro the geometric information (not all information) will be exported. Figure (46) presents the table of the exported data in "Geometry" sheet.

<table>
<thead>
<tr>
<th>Reference area</th>
<th>35.60</th>
</tr>
</thead>
<tbody>
<tr>
<td>Taper ratio</td>
<td>0.72</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>10.14</td>
</tr>
<tr>
<td>Mean Aerodynamic Chord</td>
<td>1.49</td>
</tr>
<tr>
<td>MAC (Y) location span direction</td>
<td>4.50</td>
</tr>
<tr>
<td>Sweep angle at 0.25 chord</td>
<td>3.20</td>
</tr>
<tr>
<td>Wing Aerodynamic center</td>
<td>5.27</td>
</tr>
<tr>
<td>Sweep angle LE</td>
<td>10.08</td>
</tr>
<tr>
<td>Span</td>
<td>19.00</td>
</tr>
<tr>
<td>Root Chord</td>
<td>1.57</td>
</tr>
<tr>
<td>Tip Chord</td>
<td>1.57</td>
</tr>
<tr>
<td>average chord</td>
<td>1.87</td>
</tr>
<tr>
<td>Sweep angle 0.5 chord</td>
<td>8.32</td>
</tr>
<tr>
<td>Setting angle</td>
<td>2.58</td>
</tr>
</tbody>
</table>

Figure 46: The exported wing’s geometric information table to the “Geometry” sheet

The "Wing-Export" macro has beside the previews function other functions which are plotting the three views of the wing. Figures (47), (48) and (49) show examples of the wing side view, front view and the upper view respectively.

Figure 47: An example of the wing side-view plot in "Geometry" sheet

Figure 48: An example of the wing front-view plot in "Geometry" sheet
Figure 49: An example of the wing upper-view plot in "Geometry" sheet

At can be noticed that the setting and dihedral angles are considered in the side and front view respectively. Table 18 describes the used VBA scripts in "Wing-Anatomy" sheet except the wing layout section.
Table 18: The used VBA scripts in "Wing-Anatomy" sheet except the wing layout section

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmosphere-Fuction&quot; Module</td>
<td>Calculate the atmospheric properties</td>
</tr>
</tbody>
</table>
| Macro   | "CLmax-mehtod"    | "CLmax-Method" Module       | 1. Modify the CLmax section by hiding and showing the rows depending on the chosen method.  
|         |                    |                             | 2. Calculate the leading edge parameter.                                 |
| Macro   | "Lift-Distribution" | "Lift-Distribution" Module | 1. Calculate the lift coefficient distribution over the half wing span.  
|         |                    |                             | 2. Calculate the lift coefficient at cruise condition.                  |
|         |                    |                             | 3. Calculate the lift curve slope and the lift coefficient at zero angle of attack. |
| Macro   | "Oswald"          | "Oswald-Method" Module      | Modify the Oswald’s span efficiency section by hiding and showing the rows |
| Macro   | "Wetted-Area"     | "Wing-Wetted-Area" Module   | Calculate the wing wetted area and the inner volume.                     |
| Macro   | "Lift-Distr-Figure" | "Lift-Distribution-Figure" Module | Generate the lift distribution plot.                                    |
| Macro   | "Wing-Export"     | "Export-Wing" Module        | 1. Export the wing and its geometrical data from "Wing" sheet to "Geometry" sheet.  
|         |                    |                             | 2. Create the plots of the wing 3 views.                               |
4.6 Tail

The tail geometry is defined in "Geometry" sheet. This sheet is divided for three main parts. The first part is the wing part which was discussed before in section (4.5.5) in this report. The second part is the Tail part and is considered in this section. while the last part is the fuselage part and will be discussed in the next section.

In the tail part not just its geometry is defined, also the aerodynamic coefficients, the wetted area and the inner volume are calculated.

4.6.1 Tail Geometry

To defined the tail geometry and dimensions the following data must be determined:

1. The tail configuration. The program gives the user the ability to choose between the T-tail, V-tail, cruciform and conventional tail configurations. This is done in the program by choosing a value from a drop-down list. By finishing this step, the button next to the input-cell must be run to modify "Geometry" sheet according to the chosen configuration.

2. The tail volume coefficients and arms. The program provides an on-line link to help the user to determine the tail volume coefficients. The link includes Table (8). The program uses the three presented method in section (3.5.1) to recommend three values for the tail arm. If the tail is conventional tail the user must choose two value one for the vertical stabilizer arm and the second to the horizontal stabilizer arm. For the other configurations (T-tail, V-tail and cruciform tail), just one value must be chosen which is the tail arm since the horizontal and vertical stabilizer are connected.

3. The general geometric information which means the taper ratio, aspect ratio, leading edge sweep angle and the position in the z-direction for both stabilizers. See Figure (50). The figure shows that there is an extra input-cell in the horizontal stabilizer section which is the tail setting angle. The showed tables in Figure (50) are for the conventional tail configuration. These tables will be modified if the tail is T-tail or cruciform tail. For the T-tail case, the input-cell for the location in the z-direction for the horizontal stabilizer will be hidden. While, if the tail is a cruciform tail, an extra input-cell will appear. This input-cell is used to determine the location of the horizontal stabilizer on the vertical one as a ratio to the vertical stabilizer span.

![Figure 50: General information tables for the conventional tail](image-url)
If the tail is a V-tail, the used table to defined the general geometric information is shown in Figure (51). It can be noticed that the table include an input-cell to determine the dihedral angle.

<table>
<thead>
<tr>
<th>Upper Ratio</th>
<th>V-Tail</th>
<th>Aspect Ratio</th>
<th>Location at Z direction</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.72</td>
<td></td>
<td>3</td>
<td>1</td>
</tr>
<tr>
<td>10</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>10</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Figure 51: General information table for the V-tail

By ending this step the tail geometry is defined and it is ready to be plotted. The tail will be plotted in Figures (47) and (49) by running a macro. The results are shown in Figures (52) and (53). The macro also calculates the detailed information for both horizontal and vertical stabilizers. The results are presented in the program by using two tables (one for the vertical stabilizer and one for the horizontal one) like the table in Figure (37).

Figure 52: An example of the side view of the wing and the tail together

A special case, if the tail is a V-tail then the dihedral angle will be considered during the plot. Figure (54) shows an example for the side view when the tail is a V-tail. To present the real dimensions, the program plots the upper view of the tail without considering the dihedral angle in a separate figure, as in Figure 55.

Figure 53: An example of the upper view of the wing and the tail together

Figure 54: An example for the side view when the tail is a V-tail
### 4.6.2 Tail Aerodynamic Coefficients and Wetted Area

Equations (86), (87) and (77) are used to calculate the lift curve slope, the lift coefficient at zero angle of attack and the moment curve slope respectively. The program ignore that a part of the horizontal stabilizer is inside the fuselage when it calculates the tail wetted area. Therefore the used equation to calculate the wetted area is Equation (79). To calculate the inner volume, Equation (78) is used.

Table 19 shows the used VBA script in the tail section and their functions.

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmospher-Fuction&quot; Module</td>
<td>Calculate the atmospheric properties</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Tail Section&quot;</td>
<td>&quot;Tail&quot; Module</td>
<td>1. Modify the tail section by hiding and showing the rows depending on the tail configuration.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;H-Tail&quot;</td>
<td>&quot;Tail&quot; Module</td>
<td>1. Calculate the horizontal stabilizer detailed geometric information. 2. Plot the horizontal stabilizer.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;V-T&quot;</td>
<td>&quot;V-Tail&quot; Module</td>
<td>1. Calculate the vertical stabilizer detailed geometric information. 2. Plot the vertical stabilizer.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;V-Shape-T&quot;</td>
<td>&quot;V-Shape-Tail&quot; Module</td>
<td>1. Calculate the V-tail detailed geometric information. 2. Generate all the V-tail Plots</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Tail-Method-Size&quot;</td>
<td>&quot;Tail-Method&quot; Module</td>
<td>Recommend three values for the tail arm.</td>
</tr>
</tbody>
</table>
4.7 Fuselage

The fuselage section in "Geometry" sheet is divided into two subsections.

4.7.1 Fuselage Geometry

The fuselage must be divided into a number of segments. Depending on this number, the program by running a macro generates a table to determine the segments’ main information, as in Figure [56].

![Figure 56: The generated table to determine the segments’ main information](image)

It can be notice from the figure that four values are required for each segment.

1. The number of points which are needed to define the segment in the program. This number can be two or three points. actually the program considers two points in the calculation while the third point (the middle point) is used to give the segment better shape. To clarify what do the points refer to. See Figure [57]. The part of the fuselage inside the red rectangular considered as a segments.

![Figure 57: Fuselage shape with clarifying what do the segment’s point refer to.](image)

2. The segment length. See Figure [57].

3. The segment shape. The user must choose a value from a drop-down list which includes the presented shape in Figure [19].

4. The segment’s cross section shape. The user must choose a value from a drop-down list which includes the presented cross sections in Figure [18].

After completing the segments’ main information, the detail information must be determine. The program generates a table for each segment. Figure [58] presents an example of this detail information tables. The numbers of rows in the detail information table depends on the selected number of point in the main information table. Figure [58] shows, that for each point the program requires six values.
1. The point location in (x) direction according to a fix coordinator system on the aircraft nose. See Figure (59).

2. upper view is the point location in (y) direction. The program assumes that the fuselage is symmetry according to the (xz) plane.

3. The point upper location in (z) direction (side view up). See Figure (57).

4. The point lower location in (z) direction (side view down). See Figure (57).

5. The corner radius. This value is ignored if the cross section is elliptic.

6. The circle radius. This value is required only for the half-half cross section.

By completing this step the fuselage will be defined in the program and ready to be plotted. The program plots the fuselage side and upper views as it is shown in Figure (60) and Figure (61).
4.7.2 Wetted Areas and Volume

This subsection contains two tables. The first table includes the fuselage cross sections area and perimeter at different points (the used points to define the segments), while, the second table include the segments volumes and wetted area. A macro with name "Fuselage-Wetted" is used to generate the tables and to make the calculation depending on the equations from appendix (A). This subsection also includes the total volume value, the total wetted area, the average depth and the maximum cross section.
Table (20) presents the used VBA scripts in the fuselage section, their locations and their functions.

Table 20: The Used VBA Scripts in the fuselage section and their functions

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;Rectangular&quot;</td>
<td>&quot;Area-Perimeter-Function&quot; Module</td>
<td>Calculating the area and the perimeter for the rectangular cross-section.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Elliptical&quot;</td>
<td>&quot;Area-Perimeter-Function&quot; Module</td>
<td>Calculating the area and the perimeter for the elliptical cross-section.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Half-Half&quot;</td>
<td>&quot;Area-Perimeter-Function&quot; Module</td>
<td>Calculating the area and the perimeter for the half-half cross-section.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;ShapeA&quot;</td>
<td>&quot;Area-Side-Volume-Function&quot; Module</td>
<td>Calculating the side area and the volume if the shape of the segment is the shape A.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;ShapeB&quot;</td>
<td>&quot;Area-Side-Volume-Function&quot; Module</td>
<td>Calculating the side area and the volume if the shape of the segment is the shape B.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;ShapeC&quot;</td>
<td>&quot;Tail-Method&quot; Module</td>
<td>Calculating the side area and the volume if the shape of the segment is the shape C.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Fuselage-Section-Generate&quot;</td>
<td>&quot;Fuselage-Section-Generate&quot; Module</td>
<td>Generating the segments’ main information table, Figure 56.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Fuselage-generate-rows-plot&quot;</td>
<td>&quot;Fuselage-generate-rows-plot&quot; Module</td>
<td>Generating the segment’s detail information table, Figure 58.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Fuselage-plot-table&quot;</td>
<td>&quot;Fuselage-plot-table&quot; Module</td>
<td>Generating the plot table.</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Fuselage-Plot&quot;</td>
<td>&quot;Fuselage-Plot&quot; Module</td>
<td>Plot the fuselage.</td>
</tr>
</tbody>
</table>
| Macro    | "Fuselage-wetted" | "Fuselage-Wetted-Area" Module | 1. Calculating the segments’ wetted areas and volumes.  
                                        2. Calculating the cross-section area and perimeter.  
                                        3. Calculating the average depth, the maximum cross section, the total volume and the total wetted area. |
4.8 Drag Model

The program gives the user the ability to choose the desired option for the following items:

1. The used equations to calculate the drag. The available equations are Raymer, Nicolai, Shevell, Torenbeek and Jenkinon equations.

2. The altitude and velocity since the drag depends on both.

3. Flow type which can be a laminar, mixed, turbulent or turbulent-compress flow. The turbulent-compress flow means that the compressibility effect is considered in the equations.

4. The surface type which determines the skin roughness values. See Table [23] at appendix (B).

5. The used method to calculate the induced drag. The user can choose between simplified or adjusted method.

A separate sheet in the program is used to make the aircraft drag model calculations. "Drag-Model" sheet is divided into six section. The first section is "Main Information" section. In this section, the general information is determined like the reference area and the desired flight conditions. The four following sections are used to calculate the drag coefficients for the aircraft components (wing, fuselage, horizontal stabilizer and vertical stabilizer). Each section includes the component’s geometric information and five input-cells. Three of these cells input is a value from a drop-down list to determine the skin type, the used equation and the flow type. The last two input-cells are used to determine the transition points if the flow is mixed. The last section is the induced drag section. As it was mentioned before the user can choose the used method. In addition, to calculating the induced drag coefficient, The total drag coefficient is calculated in this section for different lift coefficient. The results were presented by using a figure and a table. Figure [62] shows the drag polar plot from the program.

![Figure 62: Drag polar plot from the program](image)
The used VBA scripts in "Drag-Model" sheet are presented in Table 21.

### Table 21: The used VBA script in the "Drag-Model" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;Form-Fac&quot;</td>
<td>&quot;Form-Factor-Wing-Function&quot; Module</td>
<td>Calculate the form factor for the wing and similar surfaces.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Form-Fac-Fusleage&quot;</td>
<td>&quot;Form-Factor-Fuselage-Function&quot; Module</td>
<td>Calculate the form factor for the fuselage.</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Skin-Fric&quot;</td>
<td>&quot;Cf-coefficient-function&quot; Module</td>
<td>Calculate the skin friction coefficient</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmosphere-Function&quot; Module</td>
<td>Calculate the atmospheric properties.</td>
</tr>
<tr>
<td>Sheet script</td>
<td>-</td>
<td>&quot;Drag-Model&quot; Sheet</td>
<td>1. Determine the component’s skin roughness.</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>2. Control some inputs-cells and sections.</td>
</tr>
</tbody>
</table>
4.9 Component Weigh Estimation

Each equation from appendix (C) was converted to a VBA function. These functions are called at "Weight-Break-Down" sheet in the program. The user has the possibility to use the equations to estimate the components weights or to ignore them and insert his own values. Figure (63) presents the first part of the sheet which is used to determine if the user want to use the statistical equations to estimate the weight or not.

<table>
<thead>
<tr>
<th>Component List</th>
<th>Wing</th>
<th>H-Stabilizer</th>
<th>V-Stabilizer</th>
<th>Fuselage</th>
<th>Landing Gear Main</th>
<th>Landing Gear Nose</th>
<th>Fuel</th>
</tr>
</thead>
<tbody>
<tr>
<td>Estimate</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
</tr>
</tbody>
</table>

Figure 63: The used section to determine if the user want to use the statistical equations

"Weight-Break-Down" sheet includes also the fuel weight calculation, the engines weight, the passenger weight and payload weight. The weight of the last three items are defined in "Propulsion","Sizing" and "Sizing" sheets respectively. The used method and equations to calculate fuel weight are the used method in the initial weight estimation, section (3.1). The fuel fraction is calculated for the cruise and the liter segments by using the Equations (5) and (6) respectively. For the warm-up, climb and landing the values from Table (4) are used. If the aircraft is an electric aircraft, Equation (4) will be used. The program give the user the possibility to add an extra weight to the total weight and to insert an error ratio.

By determining the aircraft total weight, an iterative process must be started. The author suggests a step before the iterative process in order to reduce number of iterations. It is known that all the components weights estimation calculations depend on the total weight which was estimated at the beginning. In the ideal case, the calculated total weight will be equal to the estimated value, but this is not the usual case. If the user can determine the total weight which by using it in the equations the calculated total weight will be equal to it, the number of iteration will be reduced. The program is able to generate a plot as in Figure (64). The blue line in the figure presents the expected weight values (the used values in the equations) and the orange line presents the calculated total weight values. The intersection between these two lines is the value which must be used in the iterative progress to reduce the number of iterations.

The V-tail is handled as a horizontal stabilizer in the program. In this case the vertical stabilizer weight will be equal to zero. Table (22) present the used VBA scripts in "Weight-Break-Down" sheet.
Table 22: The used VBA scripts in "Weight-Break-Down" sheet

<table>
<thead>
<tr>
<th>Kind</th>
<th>Name</th>
<th>Location</th>
<th>Function</th>
</tr>
</thead>
<tbody>
<tr>
<td>Function</td>
<td>&quot;Weight-Wing1&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the wing weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Weight-HT&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the horizontal stabilizer weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Weight-VT&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the vertical stabilizer weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Weight-Fuselage&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the fuselage weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Weight-Main-Gear&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the main landing gear weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;Weight-Nose-Gear&quot;</td>
<td>&quot;Weight-Component-Estimation&quot; Module</td>
<td>Estimate the nose landing gear weight</td>
</tr>
<tr>
<td>Function</td>
<td>&quot;AtmosProp&quot;</td>
<td>&quot;Atmospher-Fuction&quot; Module</td>
<td>Calculate the atmospheric properites</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;Weight-Iteration&quot;</td>
<td>&quot;Iteration-Weight&quot; Module</td>
<td>Generate Figure 64</td>
</tr>
<tr>
<td>Macro</td>
<td>&quot;fuel-segment&quot;</td>
<td>&quot;Iteration-Weight&quot; Module</td>
<td>Control the fuel weight estimation section</td>
</tr>
<tr>
<td>Sheet script</td>
<td></td>
<td>&quot;Weight-Break-Down&quot; Sheet</td>
<td>Give the user the ability to use the equations or to insert his one value and other small functions.</td>
</tr>
</tbody>
</table>

Figure 64: The iterative plot in "Weight-Break-Down" sheet
Building an aircraft conceptual design tool within four months has been a challenge, and requires hard work to be managed. Besides the short time the author faced three main challenges during the work.

The first challenge was finding good references and combine between these references without any contradiction. This required deep understanding of the available methods and evaluating these methods against each other and against the available data. This process is necessary and useful, but it is highly time-consuming. At the end, the author decided to depend mainly on Gudmundsson [2] due to the following points:

1. Most of the time, the book includes several methods from different references for every calculation as well as a comparison between these methods to recommend one of them.

2. The used equations and methods are explained in a good way. When on the other side, some references present the final equations or suggest some values without any explanations.

3. The book uses mathematical equations and calculations more than statistics. Some other references depend mainly on statistics which is good but the mathematical equations give a better understanding.

4. The author was able to contact Dr. Gudmundsson and ask him about any doubt.

The second challenge was using Excel. Excel has many advantages since it is a common, user-friendly, nice shaped and well arranged program. It also has some negative sides. For instance, it does not support some mathematical function (for instance, the integration). Another example is that some of its functions cannot be used in the VBA script. However, still Excel’s positive sides are more than its negative sides.

The third challenge has been the currently used tool for the aircraft conceptual design at Linköping university BeX. It is a great tool and gives a good result. But at the same time, it is a complex and difficult to understand tool, without good documentation. Even in some cases when the used equations and methods are known, there is no way to figure out how they were implemented.

The new program covers a good number of configurations but still has many limitations. The most important limits are that:

1. The program can not be used to design the aircraft with payload drop.

2. The program doesn’t support the forward swept wing.

3. Just the monoplane wing can be designed and studied.
4. The tail shape is always trapezoidal and the program covers just four tail configurations, conventional tail, T-tail, V-tail and cruciform tail.

The program is not completed and needs more development. The main missing part is the performance study. Without performance study, the user can not be sure if the design fulfills the requirements or not. The program also needs to be extended to cover more configurations and concepts.

The future development should be concentrated on the following points.

1. The calculations of the empty weight CG and the CG envelope.
2. The control surfaces sizing.
3. The detail static and dynamic stability study.
4. The detailed performance study and analysis.
5. Extending the program to cover more configurations.
6 Conclusions

By the end of this thesis, all the main aims were achieved. The program covers more than 75% of the conceptual design process, as it was mentioned in the section (2.0). it gives the user the ability to design and study a number of different aircraft configurations.

The program was arranged in a simple way to make it as user-friendly as possible. For instance, different colors were used to distinguish between the various cell types. Unnecessary information is hidden and many calculations are made inside the macros and functions to give the user an easier understanding of the used methods and equations. The program is as well supported with multiple on-line links to explain several values and to prevent any possible doubts.

The naming method that was used and the fact that almost all of the cells have specific names increases the program’s robustness. The programs shape is also able to being modified depending on the information given in the input cells, thanks to the approximately twenty five macros that were built by using the VBA-scripts. This enhances the arrangement and prevents any confusion.

Future development of the program was also taken into consideration by preparing needed values.
References


A Areas and Volumes

A.1 Cross sections Areas and perimeters

Some simplification are used in this section. The circular cross section is an elliptical cross section, but its both diameters are equal. The following equations are used to calculate the cross section areas ($S_{cross}$) and perimeters ($P_{cross}$).

**Rectangular cross section:**

$$S_{cross} = W \cdot H - (0.8584 \cdot r^2)$$  \hspace{1cm} (103)

$$P_{cross} = 2(W + H) - 8 \cdot r + 2 \cdot \pi \cdot r$$  \hspace{1cm} (104)

**Elliptical cross section:**

$$S_{cross} = \pi \frac{D_1 \cdot D_2}{4}$$  \hspace{1cm} (105)

$$P_{cross} = \pi \left[ 3(a + b) - \sqrt{(3a + b)(a + 3b)} \right]$$  \hspace{1cm} (106)

**Half-half cross section:**

$$S_{cross} = W \cdot H - 0.429r^2 - 0.429 \cdot R^2$$  \hspace{1cm} (107)

$$P_{cross} = W + 2(H - R) + \pi(R + r) - 4r$$  \hspace{1cm} (108)

A.2 Fuselage sections volumes and side Areas

A.2.1 Shape (A)

Equations (109) and (110) are used to calculate the side area and the volume for this shape respectively.

$$S_{side} = P_{cross} \cdot L$$  \hspace{1cm} (109)

$$V = S_{cross} \cdot L$$  \hspace{1cm} (110)
A.2.2 Shape (B) or Paraboloid

The following mathematical expressions are used to calculate the area and the volume, from Gudmundsson [2].

The area includes the base

\[ S = \frac{\pi D}{12L^2} \left[ \left( 4L^2 + \frac{D^2}{4} \right)^{1.5} - \frac{D^3}{8} \right] \]  \hspace{1cm} (111)  

\[ V = \frac{\pi \cdot D^2 \cdot L}{8} \]  \hspace{1cm} (112)  

A.2.3 Shape (C)

Calculating the side area for this shape depends on the base cross-section. equation \[116\] is used to calculate the volume.

**Elliptical cross section**

\[ S_{side} = \frac{\pi D}{2} \left( \sqrt{L^2 + \frac{D^2}{4}} \right) \]  \hspace{1cm} (113)  

where is \( D = \sqrt{D_1 \times D_2} \)

**Rectangular cross section**

\[ S_{side} = W \left( \sqrt{L^2 + \frac{H^2}{4}} \right) + H \left( \sqrt{L^2 + \frac{W^2}{4}} \right) \]  \hspace{1cm} (114)  

**Half-Half cross section**

\[ S_{side} = \frac{W}{2} \left( \sqrt{L^2 + (H - R)^2} \right) + \frac{\pi R}{2} \left( \sqrt{L^2 + R^2} \right) + (H - R) \left( \sqrt{L^2 + \frac{W^2}{4}} \right) \]  \hspace{1cm} (115)  

**Volume**

\[ V = \frac{1}{3} S_{cross} \cdot L \]  \hspace{1cm} (116)  

where \( S_{cross} \) is the base cross section area.

A.2.4 Shape (D)

Calculating the side area for this shape depends on the base cross section.

**Elliptical cross section:** the calculations from reference [2]

\[ S_{side} = \pi \left( \frac{D_1 + D_2}{2} \sqrt{L^2 + \frac{D_1^2 - D_2^2}{4}} \right) \]  \hspace{1cm} (117)  

\[ V = \frac{\pi \cdot L}{12} (D_1^2 + D_1 \cdot D_2 + D_2^2) \]  \hspace{1cm} (118)  

84
Rectangular cross section:

\[ S_{\text{side}} = (W_1 + W_2) \times \sqrt{(H_1 - H_2)^2 + L^2} + (H_1 + H_2) \times \sqrt{(W_1 - W_2)^2 + L^2} \]  
\[ V = \frac{L}{3} (S_1 + S_2 + \sqrt{S_1 \cdot S_2}) \]  
(119)  
(120)

Half-Half cross section:

\[ S_{\text{side}} = \frac{(W_1 + W_2)}{2} \times \sqrt{(H_1 - H_2)^2 + L^2} + 0.5\pi \left( (R_1 + R_2) \sqrt{L^2 + R_1^2 - R_2^2} \right) + S_1 \]  
\[ S_1 = \frac{(H_1 - R_1) + (H_2 - R_2)}{2} \sqrt{(W_1 - W_2)^2 + L^2} \]  
\[ V = \frac{L}{3} (S_1 + S_2 + \sqrt{S_1 \cdot S_2}) \]  
(121)  
(122)  
(123)
B Drag Model

All equations and tables in this appendix from reference [2].

B.1 Skin Friction Coefficient

Thigh section includes the equations to estimate the skin friction coefficient.

1. **Laminar Flow**, Bertin [14] and Gudmundsson [2]:

\[
C_f = \frac{1.328}{\sqrt{Re}}
\]

2. **Turbulent Flow** Bertin [14] and Gudmundsson [2]:

\[
C_f = \frac{0.455}{\left(\log_{10}(Re)\right)^{2.58}}
\]


\[
C_f = \frac{0.455}{\left(\log_{10}(Re)\right)^{2.58}(1 + 0.144M^2)^{0.65}}
\]


\[
C_f = \frac{0.074}{Re^{0.2}} \left(1 - \left(\frac{X_{tr} - X_0}{C}\right)\right)^{0.8}
\]

where:

- \(Re\) = Reynolds number.
- \(M\) = Mach number.
- \(X_{tr}/C\) = the location of the transition point (ratio to the chord).
- \(X_0/C\) = the fictitious turbulent location (ratio to the chord). It is calculated by using equation().

\[
\left(\frac{X_0}{C}\right) = 36.9\left(\frac{X_{tr}}{C}\right)^{0.625} \left(\frac{1}{Re}\right)^{0.375}
\]
B.2 Skin Roughness values

Table 23 presents the roughness values for different surface types.

<table>
<thead>
<tr>
<th>Surface Type</th>
<th>Roughness</th>
</tr>
</thead>
<tbody>
<tr>
<td>Camouflage paint on aluminum</td>
<td>3.33 × 10^{-5}</td>
</tr>
<tr>
<td>Smooth paint</td>
<td>2.08 × 10^{-5}</td>
</tr>
<tr>
<td>Production sheet metal</td>
<td>1.33 × 10^{-5}</td>
</tr>
<tr>
<td>Polished sheet metal</td>
<td>0.50 × 10^{-5}</td>
</tr>
<tr>
<td>Smooth molded composite</td>
<td>0.17 × 10^{-5}</td>
</tr>
</tbody>
</table>

B.3 Form Factor

This section includes the form factor equations for different aircraft components from Gudmundsson[2].

B.3.1 Form Factor for Wing, HT, Vt, struts and Pylons

1. Torenbeek’s Equation

\[
FF = 1 + 2.7\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4
\]  

(129)

2. Shevell’s Equation

\[
FF = 1 + \frac{(2 - M^2)\cos\Lambda_{C/4}}{\sqrt{1 - M^2\cos^2\Lambda_{C/4}}} \left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4
\]  

(130)

3. Raymer’s and Nicolai’s Equation

\[
FF = \left[1 + \frac{0.6}{(x/c)_{max}} \left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right] \times \left[1.34M^{0.18}(\cos\Lambda_{tmax})^{0.28}\right]
\]  

(131)

4. JenKinson’s Equation

- Wing:

\[
FF = \left[3.3\left(\frac{t}{c}\right) - 0.008\left(\frac{t}{c}\right)^2 + 27\left(\frac{t}{c}\right)^3\right]\cos^2\Lambda_{C/2} + 1
\]  

(132)

- Tail:

\[
FF = \left[3.52\left(\frac{t}{c}\right)\right]\cos^2\Lambda_{C/2} + 1
\]  

(133)
Where:

- $\Lambda_{C/4}$ = sweep angle of the quarter chord line.
- $\Lambda_{C/2}$ = sweep angle of the mid-chord line.
- $\Lambda_{t_{\text{max}}}$ = sweep angle of the maximum thickness line.

### B.3.2 Form Factor for The Fuselage

1. **Torenbeek’s Equation**

   \[ FF = 1 + \frac{2.2}{f^{1.3}} + \frac{3.8}{f^3} \]  \hfill (134)

2. **Shevell’s Equation**

   \[ FF = 2.939 - 0.7666f + 0.1328f^2 - 0.01074f^3 + 0.0003275f^4 \]  \hfill (135)

3. **Raymer’s and Nicolai’s Equation**

   \[ FF = 1 + \frac{60}{f^3} + \frac{f}{400} \]  \hfill (136)

4. **Jenkinson’s Equation**

   \[ FF = 1 + \frac{2.2}{f^{1.3}} - \frac{0.9}{f^3} \]  \hfill (137)

Where $f$ is the fuselage fineness ratio.
C  Weight Estimation

This appendix includes the statistical equations to estimate the aircraft components weights Gudmundsson [2] and Raymer [1].

C.1  Wing Weight

\[ W_{W} = 0.036 \cdot S_{W}^{0.758} W_{FW}^{0.0035} \left( \frac{A_{RW}}{\cos^2 \Lambda_{C/4}} \right)^{0.6} q^{0.006} \lambda^{0.04} \left( \frac{100 \cdot t/c}{\cos \Lambda_{C/4}} \right)^{-0.3} (n_{z} W_{O})^{0.49} \]  

(138)

where:

- \( W_{W} \) = predicted weight of the wing in \( lb \).
- \( S_{W} \) = trapezoidal wing area in \( ft^2 \).
- \( W_{FW} \) = weight of the fuel in the wing in \( lb \).
- \( A_{RW} \) = wing aspect ratio.
- \( \Lambda_{C/4} \) = wing sweep angle at 25% of the wing mean aerodynamic chord.
- \( q \) = dynamic pressure at cruise.
- \( \lambda \) = wing taper ratio.
- \( t/c \) = wing thickness ratio to the chord.
- \( n_{z} \) = ultimate load factor.
- \( W_{O} \) = design gross weight in \( lb \).

C.2  Horizontal Tail Weight

\[ W_{HT} = 0.016(n_{z} W_{O})^{0.414} q^{0.168} S_{HT}^{0.896} \left( \frac{100 \cdot t/c}{\cos \Lambda_{HT}} \right)^{-0.12} \left( \frac{A_{RH} W_{O}}{\cos^2 \Lambda_{HT}} \right)^{0.043} \lambda_{HT}^{-0.02} \]  

(139)

Where:

- \( W_{HT} \) = predicted weight of HT in \( lb \).
- \( S_{HT} \) = trapezoidal HT area in \( ft^2 \).
- \( \Lambda_{C/4} \) = HT sweep at 25% mean aerodynamic chord.
- \( \lambda \) = HT taper ratio.
C.3 Vertical Tail weight

\[ W_{VT} = 0.073(1+0.2F_{tail})(n_zW_O)^{0.376}q^{0.122}S_{VT}^{0.873}\left(\frac{100 \cdot t/c}{\cos \Lambda_{VT}}\right)^{-0.49}\left(\frac{AR_{VT}}{\cos^2 \Lambda_{VT}}\right)^{0.357}\lambda^{0.039} \]

(140)

Where:

- \( W_{VT} \) = predicted weight of VT in \( lb_f \).
- \( F_{tail} = 0 \) for conventional tail, 1 for T-tail.
- \( S_{VT} \) = trapezoidal VT area in \( ft^2 \).
- \( \Lambda_{C/4} \) = VT sweep at 25% mean aerodynamic chord.
- \( \lambda \) = VT taper ratio.

C.4 Fuselage Weight

\[ W_{Fus} = 0.052S_{FUS}^{1.086}(n_zW_O)^{0.177}l_H^2_{HT}^{-0.051}\left(\frac{l_{FS}}{d_{FS}}\right)^{-0.072}q^{0.241} + 11.9(V_P\Delta P)^{0.271} \]

(141)

Where:

- \( W_{Fus} \) = predicted weight of the fuselage in \( lb_f \).
- \( S_{Fus} \) = fuselage wetted are in \( ft^2 \).
- \( l_{HT} \) = horizontal tail arm in \( ft \).
- \( l_{FS} \) = length of the fuselage in \( ft \).
- \( d_{FS} \) = average depth of the fuselage in \( ft \).
- \( V_P \) = volume if pressurized cabin in \( ft^3 \).
- \( \Delta P \) = cabin pressure differential in \( psi \).

C.5 Main Landing Gear

\[ W_{MLG} = 0.095(n_lW_l)^{0.768}(L_m/12)^{0.409} \]

(142)

where

- \( W_{MLG} \) = predicted weight of the main landing gear in \( lb_f \).
- \( n_l \) = ultimate landing load factor.
- \( W_l \) = design landing weight in \( lb_f \).
- \( L_m \) = length of the main landing gear in inch.
C.6 Nose Landing Gear

\[ W_{NLG} = 0.125(n_1W_t)^{0.566}(L_n/12)^{0.845} \]  \hspace{1cm} (143)

Where:

- \( W_{NLG} \) = predicted weight of the nose landing gear in \( lb_f \).
- \( L_n \) = length of the nose landing gear in inch.