Hochschule für Angewandte Wissenschaften Hamburg
Hamburg University of Applied Sciences

Master Thesis

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Reverse Engineering of Passenger Jets – Classified Design Parameters

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Abstract

This thesis explains how the classified design parameters of existing passenger jets can be determined. The classified design parameters are; the maximum lift coefficient for landing and take-off, the maximum aerodynamic efficiency and the specific fuel consumption. The entire concept is based on the preliminary sizing of jet powered civil aeroplanes. This preliminary sizing is explained in detail because it is the foundation of the final result. The preliminary sizing is combined using reverse engineering which is not a strict method. Therefore, only the basics are explained. By applying reverse engineering on the preliminary sizing and aiming for the classified design parameters as output, formulas are deviated to calculate the maximum lift coefficients, the maximum aerodynamic efficiency and the specific fuel consumption. The goal is to calculate these parameters, using only aircraft specifications that are made public by the manufacturer. The calculations are complex with mutual relations, iterative processes and optimizations. Therefore, it is interesting to integrate everything in a tool. The tool is built in Microsoft Excel and explained in detail adding operating instructions. The program is executed for miscellaneous aeroplanes, supported with the necessary comments. Investigated aeroplanes are: Caravelle 10B (Sud-Aviation), Boeing 707-320C, BAe 146-200 (British Aerospance), A320-200 (Airbus), 'Rebel' (based on A320), Boeing SUGAR High, Boeing 747-400, Blended Wing Body VELA 2 (VELA) and Dassault Falcon 8X.

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DEPARTMENT OF AUTOMOTIVE AND AERONAUTICAL ENGINEERING

Reverse Engineering of Passenger Jets – Classified Design Parameters

Task for a *Master thesis* according to university regulations

Background

For competitive reasons manufacturers try to protect their product design with its inherent parameters. This is done to protect company know-how and to maintain a possible design advantage with respect to competing products. This principle is followed not only in case of military aircraft, but also for civil passenger jets. Parameters like maximum take-off mass are known as part of the certification process. Further parameters may be given, because they are uncritical or needed for aircraft operation. Other parameters like aerodynamic efficiency or en-gine efficiency are classified information. It would be beneficial to know such parameters to do own flight performance calculations or even redo a preliminary sizing of the aircraft under in-vestigation. This can be done out of interest, educational exercise or for a more in depth case study. Knowing classified parameters would enable a comparison of various similar contempo-rary aircraft or to investigate the evolution of aircraft with their parameters throughout aviation history. Reverse Engineering is a legal possibility to acquire the knowledge withheld.

Task

The task of this thesis is to investigate classified design parameters and find a method to calcu-late them from known aircraft parameters. This should be done by preliminary aircraft sizing in combination with reverse engineering. The tasks of the project are as follows:

- Review the basics of preliminary sizing of large (CS-25) passenger jet aircraft.
- Examine the concept of reverse engineering.
- Apply reverse engineering to preliminary sizing of large passenger jet aircraft.
- Construct a tool to enable reverse engineering to the given domain and level of detail.

- Apply the tool to a number of interesting passenger jets.
- Analyze and interpret the results. Summarize the results in a case study for every investigated and reverse engineered aircraft. Compare the results of the case studies.

The report has to be written in English based on German or international standards on report writing.

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Nomenclature

A	Aspect ratio
A_{eff}	Effective aspect ratio
a	Speed of sound
B_s	Breguet factor, distance
B_t	Breguet factor, time
b	Span
C_D	Drag coefficient
$C_{D,i}$	Induced drag coefficient
$C_{D,0}$	Drag coefficient at zero lift
C_L	Lift coefficient
$C_{L,md}$	Lift coefficient for minimum drag
c_{MAC}	Mean aerodynamic chord length
$\overline{c_f}$	Skin friction factor
$(c_{L,max})_{base}$	The maximum lift coefficient of the base of an airoil.
E	Aerodynamic efficiency
e	Oswald's span efficiency factor
g	Gravitational acceleration (9,81 m/s^2)
h	Altitude
k_{APP}	Factor for approach
k_E	Factor for aerodynamic efficiency
$k_{e,NP}$	Factor span efficiency for non-planar configurations
$k_{e,WL}$	Factor span efficiency for winglet
k_L	Factor for landing
k_{TO}	Factor for take-off
L	Temperature lapse rate (0,0065 K/m)
M	Mach number
M	Molar mass of dry air (0,0289644 kg/mol)
M_{ff}	Mission fuel fraction
m	Mass
m_{ML}/m_{MTO}	Relative maximum landing mass
m/S_W	Wing loading
n_E	Number of engines
p	Local atmospheric pressure
p_0	Standard atmospheric pressure at SL (101325 Pa)
R	Range
R	Universal gass constant (8,31447 J/mol/K)
Re	Reynolds number
SFC	Specific fuel consumption

S_{ref}	Reference area
S_W	Wing area
S_{wet}	Wetted area
s	Distance / length
s_{LFL}	Landing field length
s_{TOFL}	Take-off field length / reference field length
T	Thrust
T_0	Standard temperature at SL (288,15 K)
$T/(m \cdot g)$	Thrust-to-weight ratio
t	Time
t	Airfoil thickness
t/c	Relative thickness
V	Volume
V	Speed
V_{APP}	Approach speed
V_S	Stall speed
V_1	Take-off decision speed
V_2	Take-off safety speed
$x_{(y\_c),max}$	Position of maximum camber
$x_{t,max}$	Position of maximum thickness
$(y_c)_{max}/c$	Camber

Greek symbols

ΔX (DELTA)	Additional value
Δ_x (DELTA)	Correction term
Δy	Leading edge sharpness parameter
γ (gamma)	Ratio of specific heats for air (1,4)
γ_{CLB} (gamma)	Climb gradient
γ_{MA} (gamma)	Missed approach climb gradient
η (eta)	Efficiency
Λ (LAMBDA)	Sweep angle
λ (lambda)	Taper
μ (mu)	Bypass ratio
μ (mu)	Dynamic viscosity
φ (phi)	Sweep angle
π (pi)	3,141592653589793…
ρ (rho)	Density
σ (sigma)	Relative air density

Indices

CLB	Climb
CR	Cruise
DES	Descend
F	Fuel
f	Flap
H.L.	Hinge line
L	Landing
LE	Leading edge
MA	Missed approach
max	Maximum
OE	Operating empty
PL	Payload
RES	Reserve
s	Slat
TO	Take-off
W	Wing

List of Abbreviations and Acronyms

AAC	Aircraft Approach Category
ADG	Aircraft Design Group
ARC	Aerodrome Reference Code
BAe	British Aerospace
BWB	Blended Wing Body
CAD	Computer-aided Design
CFR	Code of Federal Regulations
CG	Center of Gravity
CS	Certification Specification
DATCOM	Data Compendium (USAF aerodynamics methodology report)
EASA	European Aviation Safety Agency
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulations (certification specs)
FL	Flight Level
ICAO	International Civil Aviation Organization
ISA	International Standard Atmosphere
LR	Long Range
MA	Missed Approach
MR	Medium Range
MTOW	Maximum Take-off Weight
MZFW	Maximum Zero Fuel Weight
NACA	National Advisory Committee for Aeronautics
NEO	New Engine Option
OAPR	Overall Pressure Ratio
PAX	Passenger
PW	Pratt & Whitney
SFC	Specific Fuel Consumption
SI	International System (Système Internationale)
SL	Sea Level
SR	Short Range
SUGAR	Subsonic Ultra Green Aircraft Research
RE	Reverse Engineering
TET	Turbine Entry Temperature
ULR	Ultra-Long Range
USA	United States of America
USAF	United States Air Force
VELA	Very Efficient Large Aircraft
WW I	First World War
WW II	Second World War

List of Definitions

Camber
"Camber (noun) is the degree to which an aircraft wing or other aerofoil curves up from its front edge and down again to its back edge." (**Allen 2006**)

Comprehensive
"Comprehensive (adj) means covering completely or broadly." (**Allen 2006**)

Circuitous
"Circuitous (adj) indirect in route or method; roundabout." (**Allen 2006**)

Drag
"Drag (noun) is the retarding force acting on a body, e.g. an aircraft, moving through air, water or other fluid, parallel and opposite to the direction of motion." (**Allen 2006**)

Flap
"Flap (noun) is a movable control surface on an aircraft wing for increasing lift during take-off or drag during landing." (**Allen 2006**)

Lift
"Lift (noun) is the component of the aerodynamic force acting on an aircraft or wing that is perpendicular to the relative wind and usu constitutes the upward force opposing the pull of gravity." (**Allen 2006**)

Loiter
"Loiter (adj intrans) is to remain in an area for no obious reason." (**Allen 2006**)

Matching chart
A matching chart shows the two-dimensional relation between the wing loading and the thrust-to-weight ratio for landing, take-off, second segment, cruise and missed approach.

Slat
"Slat (noun) is a control surface along the leading edge of a wing that can be extended forward to create a gap (slot) to improve airflow." (**Allen 2006**)

Turbofan
"Turbofan (noun) is a jet engine with a turbofan. It refers to the fan that is directly connected to and driven by a turbine and is used to supply air for cooling, ventiliation or combustion." (**Allen 2006**)

Verification
"Verification (noun) is the act or instance of verifying." (**Allen 2006**)

Verify
"Verify (verb trans) to ascertain the truth, accuracy, or reality of something." (**Allen 2006**)

1 Introduction

1.1 Motivation

Nowadays, flying is as common as driving a car. But it is only because of a few people, who believed humankind would be able to fly one day, that all of this was made possible. History shows some great pioneers. Ambitious founders are Otto Lilienthal, Orville and Wilbur Wright (a.k.a. the Wright brothers) and Gustav Weißkopf. These men had the courage and perseverance to try what no human ever tried before, flying. Thanks to these intelligent and creative men, humankind is now able to cover large distances through the skies.

The first flying devices came up in the last decennium of the 19th century. Many pioneers got inspired by nature. Birds and insects have been their great examples and remain important to the aircraft history. And so, the first aeroplanes started with a winglike shape, just like birds. The problems that one had to encounter were lift, stability, controllability and manoeuvrability.

The development of the aeroplanes was getting more theoretical thanks to the earlier work of Bernouilli, Leonard Euler, Ernst Mach, Ludwig Prandtl and Theodore von Kármán. Fluid dynamic theory developed, wind tunnels were built and measurements on test models were collected.

In 1914, the First World War started, followed by the Second World War from 1939 to 1945. War has always negative effects on people. But during war, technology develops at a higher tempo because of the competitive element between parties. The development of the aeroplanes in WW I and WW II has known a big and fast progress. Planes got more efficient, could carry more payload, got faster and in general, improved performance. Also the stability, control and manoeuvrability from the airplanes knew an important development in this time period. But the most important improvement was the introduction of the jet engines at the end of WW II (Messerschmitt Me 262, Germany).

When the WW was over, people came up with new possibilities for the use of aircrafts.[1] An industrial market for the (jet powered) **passenger aeroplane** arose. By flying individuals from one point to another, a new pristine market emerged. The attractiveness of the idea was that an individual could go to a far destination in a safe and fast way. The Havilland DH 106 Comet (1949, United Kingdom) and Caravelle (1955, France) are one of the first jet powered passenger aeroplanes in the world.

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1 Although there were a few precursors such as Handley Page Type W (1919), Junkers Ju 49 (1931, Germany) and Douglas DC-3 (1935, USA), which were all propeller driven aeroplanes.

Since that time, aeroplanes need to go faster, fly more efficient, be safer, more ecological and most important, fly more profitable. Aeroplanes are designed and created by humans. Just like a pocket watch, it consists out of thousands of pieces which all work perfectly together. This flawless co-operation of parts demands knowledge, knowhow, courage, intellect and lots of perseverance. But how does one **return** into the steps of the designer and figure out how every piece works correctly with the other ones?

1.2 Definitions

'Reverse Engineering of Passenger Jet Classified Design Parameters' is the title of this thesis. In this section, every term will be defined, using two descriptive English dictionaries; **Longman 2009** and **Allen 2006**.

Reverse
The term *reverse* is defined as follows (according to **Longman 2009**):

> *Reverse (verb trans) is to change something, such as a decision, judgment or process so that it is the opposite of what it was before.*

In this thesis, reverse has the meaning to change a calculation method in a way that the inputs become the outputs. Aircraft technology requires a big amount of parameters, therefore the term reverse cannot be taken literally in its meaning. Not every input becomes an output and vice versa. In this thesis, the reversing is done by aiming on specific parameters which has to become an output. All the other parameters are unchanged in there meaning and thus remain inputs.

Engineering
According to **Allen 2006**, the definition of the term *engineering* is as follows:

> *Engineering (noun) the application of science and mathematics by which the properties of matter and the sources of energy in nature are made useful to human beings in machines, structures, processes, etc.*

This definition corresponds with the context of *engineering* in this thesis. Science and mathematics that are used, are the main tool for designing an aeroplane. The engineering in this thesis is pure theoretical engineering.

Passenger
The term *passenger* is defined as follows according to **Longman 2009**:

> *A passenger (noun) is a person who travels in any vehicle (boat, aeroplane, car, etc.) but who is not the driver or anyone working there.*

A similar definition is provided by **Allen 2006**:

> *A passenger (noun) is somebody who travels in, but does not operate, a public or private conveyance.*

In the context of this thesis, both meanings of *passenger* can be taken literally. The focus lies on passenger aeroplanes only. This excludes cargo flights and military operations.

Jet
Both dictionaries (**Longman 2009** and **Allen 2006**) provide three similar definitions for the term *jet*:

1. *An aircraft powered by one or more jet engines.*
2. *A narrow stream of liquid or gas forced out under pressure from a narrow opening or a nozzle.*

The third definition is more related to the thesis. This definition refers to the term *jet engine*:

3. *Jet = jet engine (noun) pushes out a stream of hot air and gases behind it, used in an aircraft.*

These three definitions provide an accurate description of the term *jet* in the context of the thesis. It is an engine that makes use of a stream of hot air and gases, which is pushed out through a small hole, to provide an aeroplane from thrust.

Classified
According to **Allen 2006**, the term *classified* is defined as follows:

> *Classified (adj) said of information, a document, etc: withheld from general circulation for reasons of national or military security.*

Classified is in the sense of meaning for this thesis, interpreted as information that is withheld from general circulation in order to protect the studies and work of a company's design. In this work, those companies can be Airbus, Boeing, Embraer, Fokker, Mc Donnell Douglas, Bombardier, etc. To protect their design, specifications about the aeroplanes are kept classified.

Design
Allen 2006 defines *design* as a transitive verb in two different ways:

1. *Design (verb trans) to conceive and plan (something) out in the mind.*
2. *Design (verb trans) to devise (something) for a specific function or end.*

The definitions are similar; to conceive or devise something for a specific function out in the mind. The meaning can be taken literally for this thesis. Something is conceived or devised by the designers. The specific function is an aeroplane that meets the requirements.

Parameter
The term *parameter* is defined in two different ways by **Allen 2006**:

1. *Parameter (noun) an arbitrary constant whose value characterizes a member of a system, e.g. a family of curves*
2. *Parameter (noun) any of a set of physical properties whose values determine the characteristics or behaviour of something.*

The meaning of *parameter* in the second definition is most applicable for this thesis. The *parameters* which will be discussed, describe the behaviour of the aeroplane. These values are physical properties of the aeroplane.

1.3 Objective of the Thesis

An aeroplane is a technical piece of art. Everything is calculated and designed in a way that the requirements are met. So each airplane type is different, depending on the designers' views. The flying vehicle consists out of many pieces which all have to co-operate flawless in order to guarantee the **passengers'** safety. This thesis is about returning into the designers' steps and how every single piece works with the other ones. But is it possible to put a value on the way of co-operation of these pieces? Is it possible, using only aircraft physical specifications, to know what these values are?

But what are those values? What describes an aeroplane its characteristics? When this is known, one is able to investigate different aeroplanes. They can be compared with each other: old versus new models, look-a-likes, evolution of the aircraft industry or a comparison of contemporaries. The values could also be used to perform further calculations e.g. flight mechanic calculations or do a preliminary sizing or to start a redesign. The values that describe the aeroplanes' characteristics are essential and basic inputs for lots of calculations in aircraft technology.

Unfortunately, these parameters are classified by the manufacturer companies (e.g. Airbus, Boeing, Fokker, Mc Donnell Douglas, Bombardier, Dassault, Embraer or Lockheed). But is there a way to gain this information of a finished design? And if there is a way, is it possible for a user to require accurate and reliable results in a quick and user friendly way? This thesis will provide an answer for all these questions.

The aim of this thesis is to provide a tool for users that makes it possible to calculate the companies' secrets (also called values that describe the aeroplanes' characteristics) from a finished design. If the user is able to do this, a list of possibilities opens:

- Investigation of aeroplanes (old versus new, comparison of contemporaries, chart the evolution of aeroplanes, comparing the efficiency and quality through history).
- To know the basic values for further aircraft technology calculations (e.g. flight mechanics, preliminary sizing, etc).

These possibilities are the motivation for this work. The masterthesis is built on a foundation of research questions:

- Which values describe an aeroplane's characteristics?
- How does one return into the steps of the designer in order to find the mutual relation between every working piece?

The answer on these questions form the pillars of this masterthesis. The answers develop through the work in the following steps:

- Review the basics of preliminary sizing of large (CS-25) passenger jet aircraft.
- Examine the concept of reverse engineering.
- Apply reverse engineering to preliminary sizing of large passenger jet aircraft.

These pillars support a development of a working tool. This working tool is the result of the entire masterthesis and can be used by many people. The tool is built in Microsoft Excel and it has to meet certain requirements:

- Accurate results
- Reliable results
- Quick
- User friendly (e.g. provided from notifications)
- Multifunctional
- Possibility for a direct print-out of the results

Eventually, the reverse engineering program is executed on miscellaneous passenger jets. The final part of this thesis is a case study of every investigated and reverse engineered aeroplane.

1.4 Literature Review

Most of the consulted sources for this masterthesis are written in English. A few sources are written in German. No other languages are used and thus not a requirement in order to understand these sources.

Chapter 2 is the start of the thesis. It explains a possible method to perform the preliminary sizing of an aircraft. This method is based on **Loftin 1980** (assisted by **Scholz 2015**) and uses the same five subsections: landing, take-off, missed approach, second segment and cruise. This source is used to understand the philosophy of the aircraft preliminary sizing. It also contains lots of interesting graphs and empirical results. Note that only chapter 'III – Sizing of Jet Powered Cruising Aircraft' is consulted from **Loftin 1980**.

Reverse engineering is not included in a basic training program for aeronautical engineering. **Otto 2001** starts from the very beginning of the concept 'reverse engineering'. Everything is explained in detail, it is also supported with examples.

Scholz 2015 is the most used source for this masterthesis. This source explains aircraft design. One chapter is about aircraft preliminary sizing and is based on **Loftin 1980**. Therefore, it is used next to **Loftin 1980** to gain the knowledge concerning aircraft preliminary sizing. Other interesting parts are the wing design and high-lift systems. Multiple chapters explain the theoretical approach for calculating lift coefficients in a detailed way. This is used as build-up for the verification calculations.

1.5 Structure of the Work

The main goal of this thesis, is to provide a program which makes it possible to gain the classified information in a quick and accurate manner. Because it has to be quick and user friendly, it is important that the number of inputs is limited. Therefore, the calculations are only based on the preliminary sizing and not a conceptual or final design approach. For this reason, the thesis will start with the explanation of the preliminary sizing.

Chapter 2 explains the preliminary sizing for jet powered passenger aircrafts. The approach is based on **Loftin 1980**. This is divided into five subsections; landing, take-off, missed approach, second segment and cruise. Each section explains what is important and what defines the aeroplane.

Chapter 3 explains the reverse engineering. It defines reverse engineering and which steps that are necessary in order to perform the reverse engineering (**Otto 2001**). This serves as a basic method which makes it possible to reverse the preliminary sizing and end up with the classified parameters.

Chapter 4 describes the actual reverse engineering of the preliminary sizing. The same layout is handled as in Chapter 2. It explains how the formulas are originated.

Chapter 5 describes the build-up of the program. The program is an Excel file which consists of several tabs. The purpose of each tab is explained in detail. By reading this, the user is able to use the program and perform the RE on an aeroplane.

Chapter 6 this chapter is a description of the RE that is performed for several airplanes. It contains a brief description of each aircraft, followed by the results of the reverse engineering and finalised with a discussion of the findings. Some aeroplanes are compared to each other. The Excel files can be found in the appendix and could be helpful as an example for the user.

Appendix A shows the results of the program Caravelle 10B_Reverse Engineering.xlsm

Appendix B shows the results of the program Boeing 707-320C_Reverse Engineering.xlsm

Appendix C shows the results of the program BAe 146-200_Reverse Engineering.xlsm

Appendix D shows the results of the program A320-200_Reverse Engineering.xlsm

Appendix E shows the results of the program Rebel_Reverse Engineering.xlsm

Appendix F shows the results of the program Boeing SUGAR High_Reverse Engineering.xlsm

Appendix G shows the results of the program Boeing 747-400_Reverse Engineering.xlsm

Appendix H shows the results of the program BWB_Reverse Engineering.xlsm

Appendix I shows the results of the program Dassault Falcon 8X_Reverse Engineering.xlsm

2 Aircraft Preliminary Sizing

The aim of preliminary sizing is to have a fast method wherein a rough estimation can be made of the aircraft (design) parameters. Therefore the design has to meet the airport- and cruise performances. Together they are called 'the performance criteria' or 'requirements'.

Aircraft design parameters

- Take-off mass m_{TO}
- Fuel mass m_F
- Operating empty mass m_{OE}
- Wing area S_W
- Take-off thrust T_{TO}

Performance criteria / requirements

Cruise performance:

- Payload m_{PL}
- Range R
- Mach number M_{CR}

Airport performance:

- Take-off field length s_{TOFL}
- Landing field length s_{LFL}
- Climb gradient γ_{CLB}
- Missed approach climb gradient γ_{MA}

In this study, the focus lays in jet powered aircrafts which are developed to make steady cruise flights. To get a certification, the designed aircraft has to meet with the prescribed rules. These rules are imposed by the Federal Aviation Administration (FAA, United States of America) or from the European Aviation Safety Agency (EASA, Europe). The FAA developed the Federal Air Regulations (FAR) which finds its origin in the Code of Federal Regulations (CFR), in specific Title 14. The EASA developed the Certification Specification (CS) which are quite similar to the FAR. Since this study focuses only on jet powered aircrafts, two distinctions are made in the regulations which an aeroplane has to meet to obtain a certification. For light jets (weights less than 12 500 lb or 5700 kg) FAR Part 23 or CS-23 applies to obtain a certification. For large jet powered aeroplanes FAR Part 25 or CS-25 is applied. The EASA-CS-25 is applied in this case because the emphasis is placed on large aeroplanes.

The preliminary sizing consists out of five different parts: landing, take-off, second segment, missed approach and cruise. For each of them, certain input values are necessary and the aircraft design parameters are the output. Figure 2.1 is the flow diagram for preliminary sizing where the inputs and outputs for the five blocks are shown. Each block will be explained in detail.

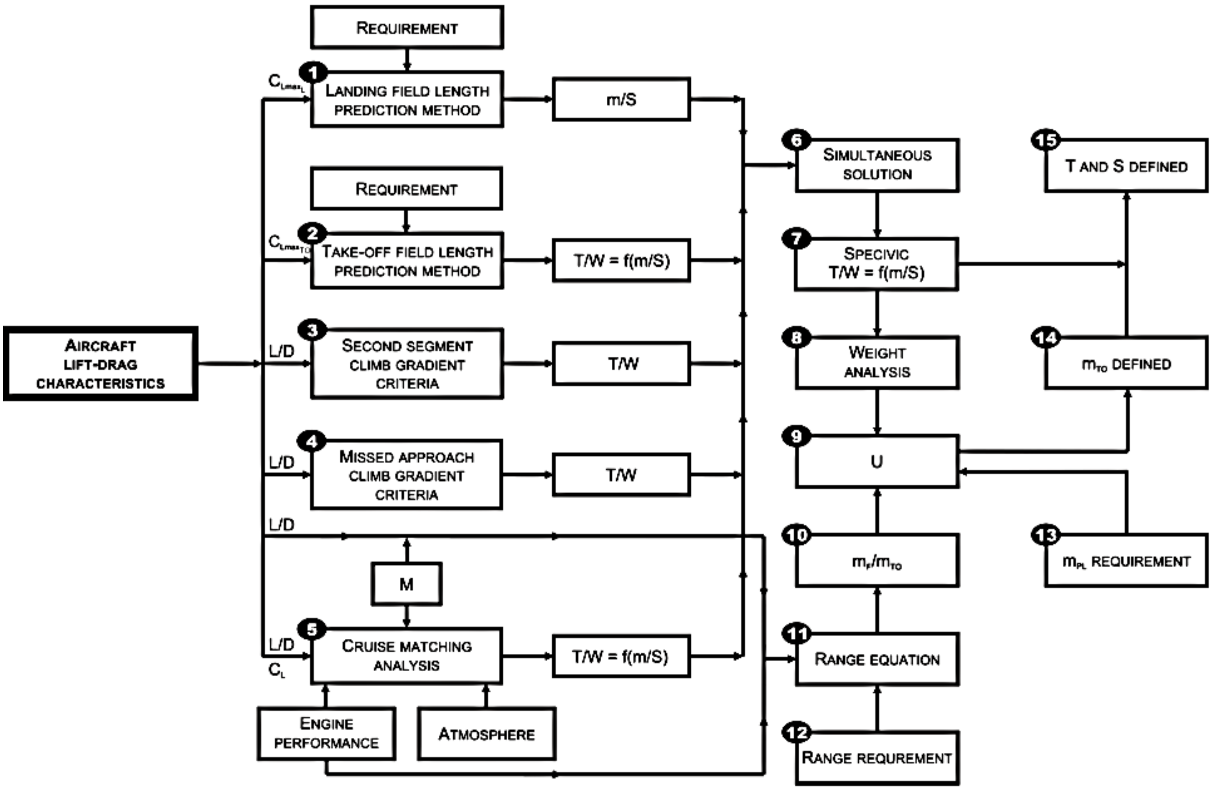


Figure 2.1 Aircraft-sizing flow diagram for preliminary sizing for jet powered aircraft (**Scholz 2015** based on **Loftin 1980**)

The method to find the thrust-to-weight ratio and the wing loading for every section will be briefly explaned, according to **Scholz 2015** and **Loftin 1980**. In the end, the relation between the thrust-to-weight ratio and the wing loading of every part will be plotted in a 'matching chart'. This chart makes it possible to visualise the design point. In the end, the aircraft design parameters are calculated according to the design point.

2.1 Landing

To summarize the rules for landing according to CS-25, it can be said that the landing distance is the horizontally distance, starting from the point at which the aircraft is 15 m (50 ft) above the ground, to the point where the aeroplane is brought to standstill on a dry runway. During the approach in steady gliding flight, the speed should not be less than 1,3 times the stalling speed.

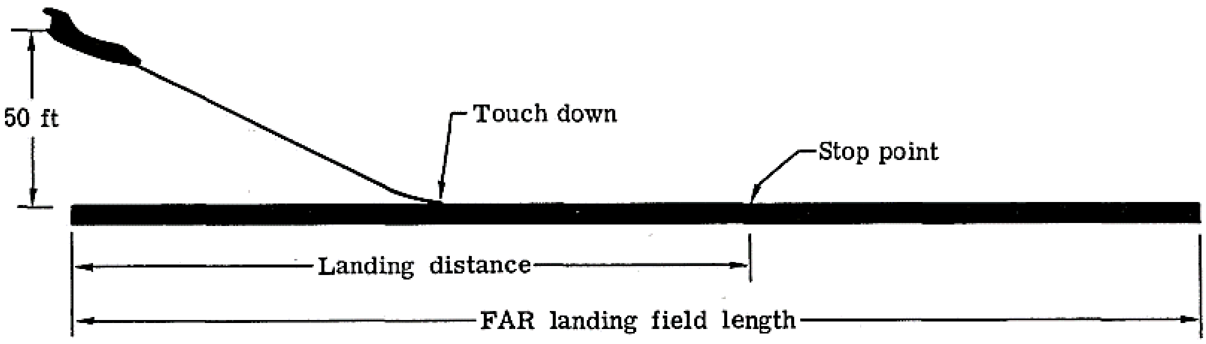


Figure 2.2 Landing field length (**Loftin 1980**)

When an aircraft lands on an airfield, the landing field length s_{LFL} must be smaller than the landing distance that is available at that airport. According to FAR and CS the landing field length has a certain maximum. This means that the wing loading can be calculated since the landing field length is a constraint. If this distance is unknown, it is possible to calculate this by using the approach speed V_{APP}. For the section landing, the wing loading is independent on the thrust-to-weight ratio. It is calculated with the following formula:

$$V_{APP} = k_{APP} \cdot \sqrt{s_{LFL}} \tag{2.1}$$

$$k_{APP} = 1{,}70 \sqrt{m/s^2}$$

$$\frac{m_{MTO}}{S_W} = k_L \cdot \sigma \cdot s_{LFL} \cdot C_{L,max,L} \cdot \frac{m_{MTO}}{m_{ML}} \tag{2.2}$$

$$k_L = 0{,}107\ kg/m^3$$

2.2 Take-off

The take-off distance is the length starting from the point where the take-off starts to the point where an aircraft has reached an altitude of 11 m (35 ft) (Figure 2.3). The take-off distance is also called the balanced field length. The take-off distance for an aircraft according to FAR includes some safety factors in case of engine failure. During acceleration on the apron, the aircraft reaches certain speeds. The first speed that has to be reached is V_1. This is the maximum speed where the pilot is able to make a safe stop on the remaining field length. If this speed is not reached and there are some malfunctions at the airplane (such as engine failure), the pilot has to stop the aircraft without take-off. V_1 is also the speed where an aircraft can reach V_2 with one engine inoperative. This means, when an aircraft has exceeded this speed and problems occur, the pilot has to take-off. In the rare case that an engine fails at the exact speed V_1, the pilot can choose what to do. The second speed is V_2, at this speed the airplane is able to make a safe take-off even with one inoperative engine.

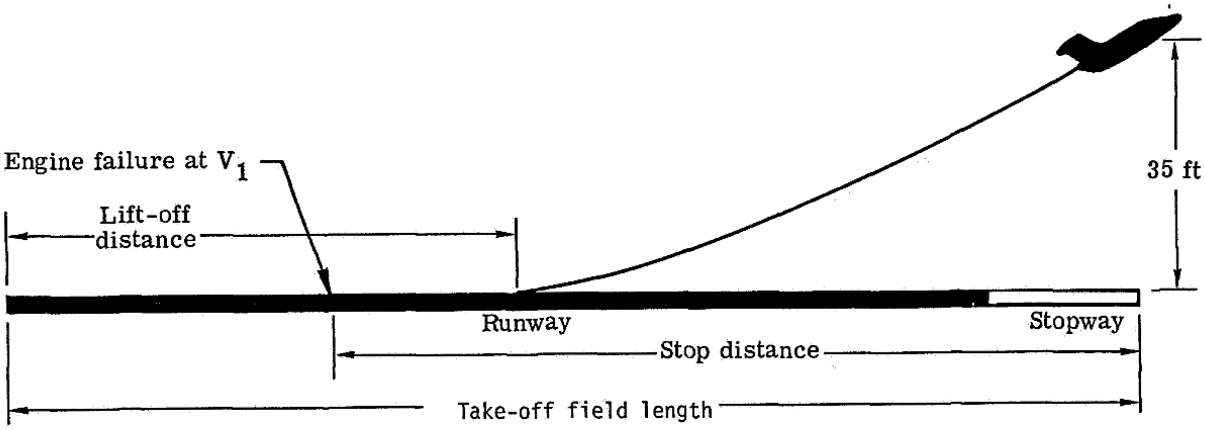


Figure 2.3 Take-off field length (**Loftin 1980**)

In order to use the take-off field length as a constraint for the preliminary sizing of an aircraft, the relation between the balanced field length and the wing loading and the thrust-to-weight ratio must be known:

$$\frac{T_{TO}}{m_{TO} \cdot g} = \frac{k_{TO}}{\sigma \cdot \; C_{L,max,TO} \cdot s_{TOFL}} \frac{m_{TO}}{S_W} = a \cdot \; \frac{m_{TO}}{S_W} \tag{2.3}$$

$$a = \frac{k_{TO}}{\sigma \cdot \; C_{L,max,TO} \cdot s_{TOFL}} \tag{2.4}$$

$$k_{TO} = 2{,}34 \; m^3/kg$$

With this formula, the thrust-to-weight ratio is described as a function of the wing-loading. And the factor ‘a’ is the slope, which is defined by the take-off constant, the weather conditions, the maximum lift coefficient in take-off configuration and the take-off field length.

2.3 Second Segment

After take-off, the take-off climb path starts. This path is divided in several segments. The first segment starts at the end of the take-off (airplane reaches an altitude of 35 ft). The second segment starts when the landing gear is fully retracted and goes to the point where the aircraft has reached an altitude of 400 ft, ready to get the flaps out of take-off configuration. In this segment, it is important that a minimum climb rate can be remained, at a speed V_2, even when there is one engine inoperative. This means that the engines of an aircraft must have a sufficient amount of thrust to meet the minimum climb rate. According to FAR, the climb gradients are as followed:

- 2,4% for two-engined aeroplanes
- 2,7% for three-engined aeroplanes
- 3,0% for four-engined aeroplanes

$$\frac{T_{TO}}{m_{MTO}\cdot g}=\frac{n_E}{n_E-1}\left(\frac{1}{E}+\sin\gamma_{CLB}\right) \tag{2.5}$$

$$E=\frac{C_{L,TO}}{C_{D,0}+\Delta C_{D,flaps}+\frac{C_{L,TO}^2}{\pi\cdot\ A\ \ e}} \tag{2.6}$$

To meet the regulations for the second segment, an aeroplane must achieve a certain climb gradient at all times, even with one failed engine. This limitation gives a value for the thrust-to-weight ratio and is independent from the wing loading.

2.4 Missed Approach

During a normal approach for landing, it can occur that for some reason the pilot can’t land the aeroplane. In this situation, the pilot has to climb again and make a turnaround. This manoeuvre must be possible with one inoperative engine. Because the aircraft is in the landing configuration, a lot of drag has to be overcome due to the flaps (which are in landing position) and an extended landing gear. According to FAR, the climb gradient, depending on the number of engines, must be as followed:

- 2,1% for two-engined aeroplanes
- 2,4% for three-engined aeroplanes
- 2,7% for four-engined aeroplanes

The regulations demand sufficient thrust in an aircraft to perform this manoeuvre (with one engine inoperative).

$$\frac{T_{TO}}{m_{MTO}\cdot g}=\frac{n_E}{n_E-1}\left(\frac{1}{E}+\sin\gamma_{MA}\right)\left(\frac{m_{ML}}{m_{MTO}}\right) \tag{2.7}$$

$$E=\frac{C_{L,L}}{C_{D,0}+\Delta C_{D,flaps}+\Delta C_{D,slat}+\Delta C_{D,gear}+\frac{C_{L,L}^2}{\pi\cdot\ A\ \ e}} \tag{2.8}$$

$$C_{L,L} = C_{L,max,TO} \left(\frac{V_S}{V}\right)^2 = \frac{C_{L,max,TO}}{1{,}3^2} \tag{2.9}$$

The formula is quite similar to the calculations for the second segment. The difference is that the aircraft has now the maximum landing mass and that the aerodynamic efficiency is influenced by the increase of drag and the lift coefficient for landing. The increase of drag is caused by the flaps which are in landing configuration and the extended landing gear.

2.5 Cruise

The following method will describe how the cruising criterion can be determined. The cruising criterion includes that the aerodynamics from an aeroplane and the characteristics of an engine are adjusted in a way that, for a given range and Mach number, a minimum amount of fuel is necessary. The method consist out of two parts: thrust-to-weight ratio and wing loading. Eventually these parts will be combined to use later on in the matching chart.

2.5.1 Thrust-to-Weight Ratio

An aircraft is designed for a specific range. Since the travelled distance mainly happens during cruise, the range has to be defined by the characteristics which match the airplane in cruise configuration. The relation between the range, the cruising speed and the characteristics of the airplane and engines, are given by the Breguet range equation for jet powered aircrafts:

$$R = \frac{V_{CR} \cdot E}{SFC} \cdot ln\left(\frac{1}{1 - \frac{m_F}{m}}\right) \tag{2.10}$$

For a flight mission it is important that the relative fuel mass m_f/m is as low as possible to reduce the necessary amount of fuel and to reduce the total weight of an aircraft, or to increase the payload and thus making more economic profit. The Breguet factor is inversely proportionate to the fuel fraction. This means that an aeroplane has to fly at the maximum aerodynamic efficiency during cruise. This is not always possible since it depends on the speed. When one aimes to fly at maximum aerodynamic efficiency, you need to fly at the minimum drag speed. When one wants to fly in such a way that the maximum range is obtained, you fly at $\sqrt[4]{3}$ times the minimum drag speed. This changes the aerodynamic efficiency. The actual aerodynamic efficiency becomes:

$$E = \frac{2E_{max}}{\frac{1}{\frac{C_L}{C_{L,md}}} + \frac{C_L}{C_{L,md}}} \tag{2.11}$$

$$C_L = \frac{C_{L,md}}{\left(\frac{V}{V_{md}}\right)^2} \tag{2.12}$$

$$E_{max} = k_E \sqrt{\frac{A}{S_{wet}/S_W}} \tag{2.13}$$

$$k_E = \frac{1}{2} \sqrt{\frac{\pi \cdot e}{\overline{c_f}}} \tag{2.14}$$

Loftin 1980 $k_E = 14{,}9$

Raymer 1989 $k_E = 15{,}8$

The actual aerodynamic efficiency is dependent on the maximum aerodynamic efficiency. This value is determined theoretically with equation (2.13). This relation is deviated by plotting points of real aircrafts (Figure 2.4).

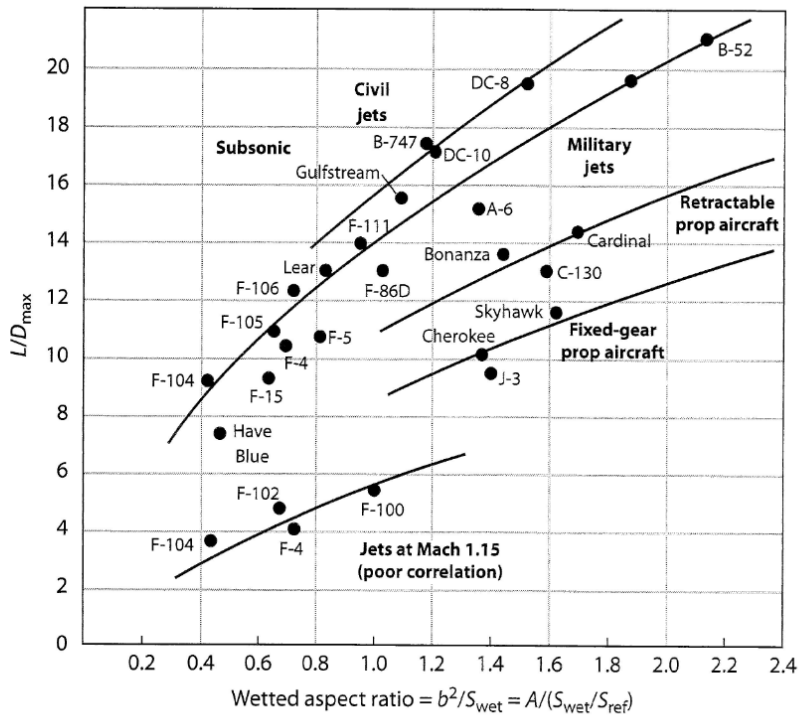


Figure 2.4 Maximum aerodynamic efficiency trends (**Raymer 2012**)

To find the relative wetted area, Figure 2.5 is used. A typical value for jet powered passenger aeroplanes is a value between 6,0 and 6,2.

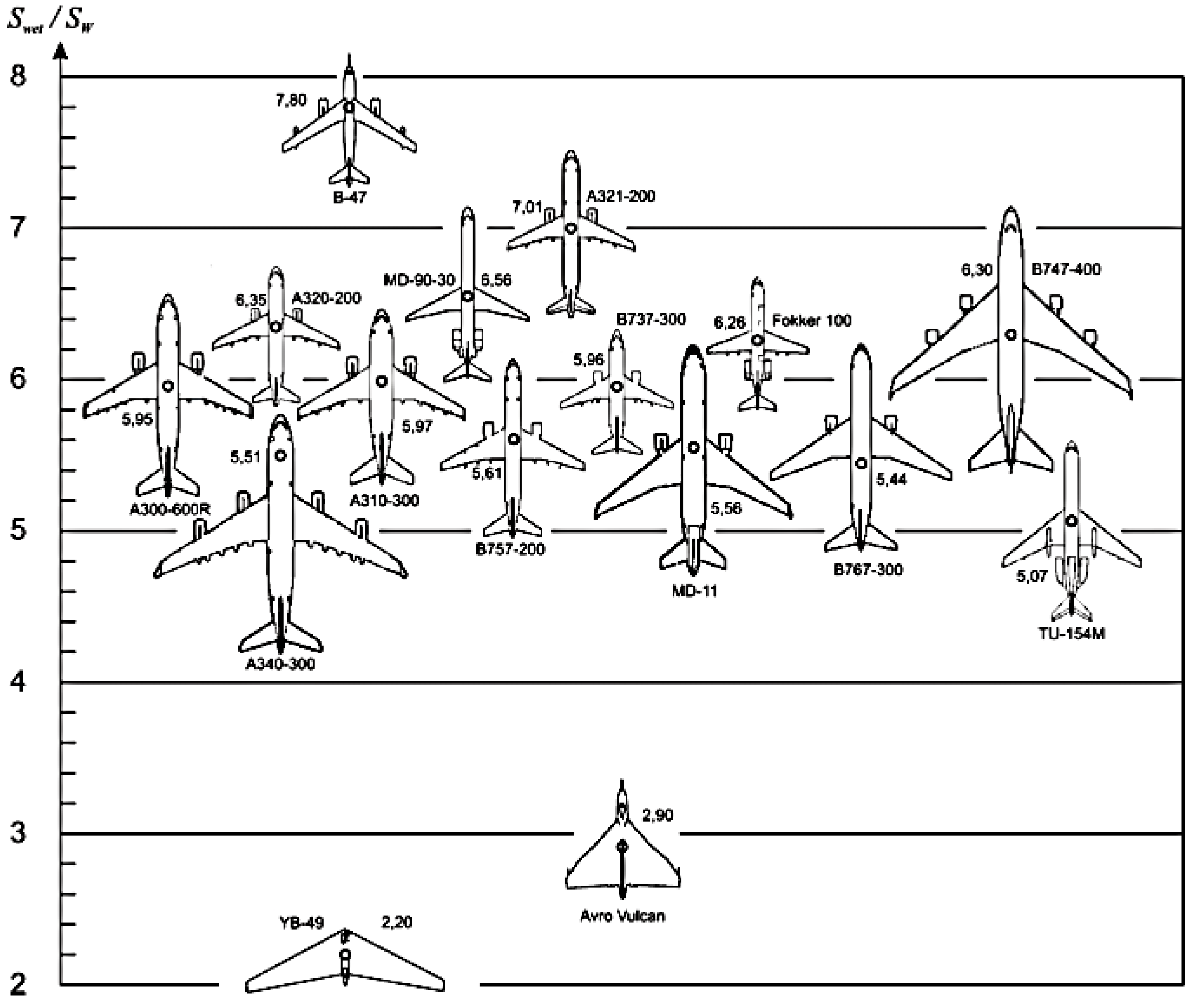


Figure 2.5 Aircraft plan forms and their relative wetted area S_{wet}/S_W (based on **Raymer 1989**)

Figure 2.1 shows for cruise, that the aerodynamic efficiency and the lift coefficient is required to find the thrust-to-weigh ratio as a function of the wing loading. The thrust-to-weight ratio is calculated with the following formula:

$$\frac{T_{MTO}}{m_{MTO} \cdot g} = \frac{1}{\left(\frac{T_{CR}}{T_{TO}}\right) \cdot E} \tag{2.15}$$

The quotient of the cruise thrust and the take-off thrust can be found in engine diagrams when the cruise Mach number and the cruise altitude is known. Figure 2.6 shows the thrust relationship for a normal jet powered airplane. The diagram clearly shows that the ratio is approximately constant for Mach numbers which are bigger than 0,8. Another way to find this ratio is to use this constant relation and calculate the ratio with formula (2.16). This formula can be used for every jet powered aeroplane cruising at Mach 0,8 and more.

$$\frac{T_{CR}}{T_{TO}} = (0{,}0013\mu - 0{,}0397)\frac{1}{km} \cdot h_{CR} - 0{,}0248\mu + 0{,}7125 \tag{2.16}$$

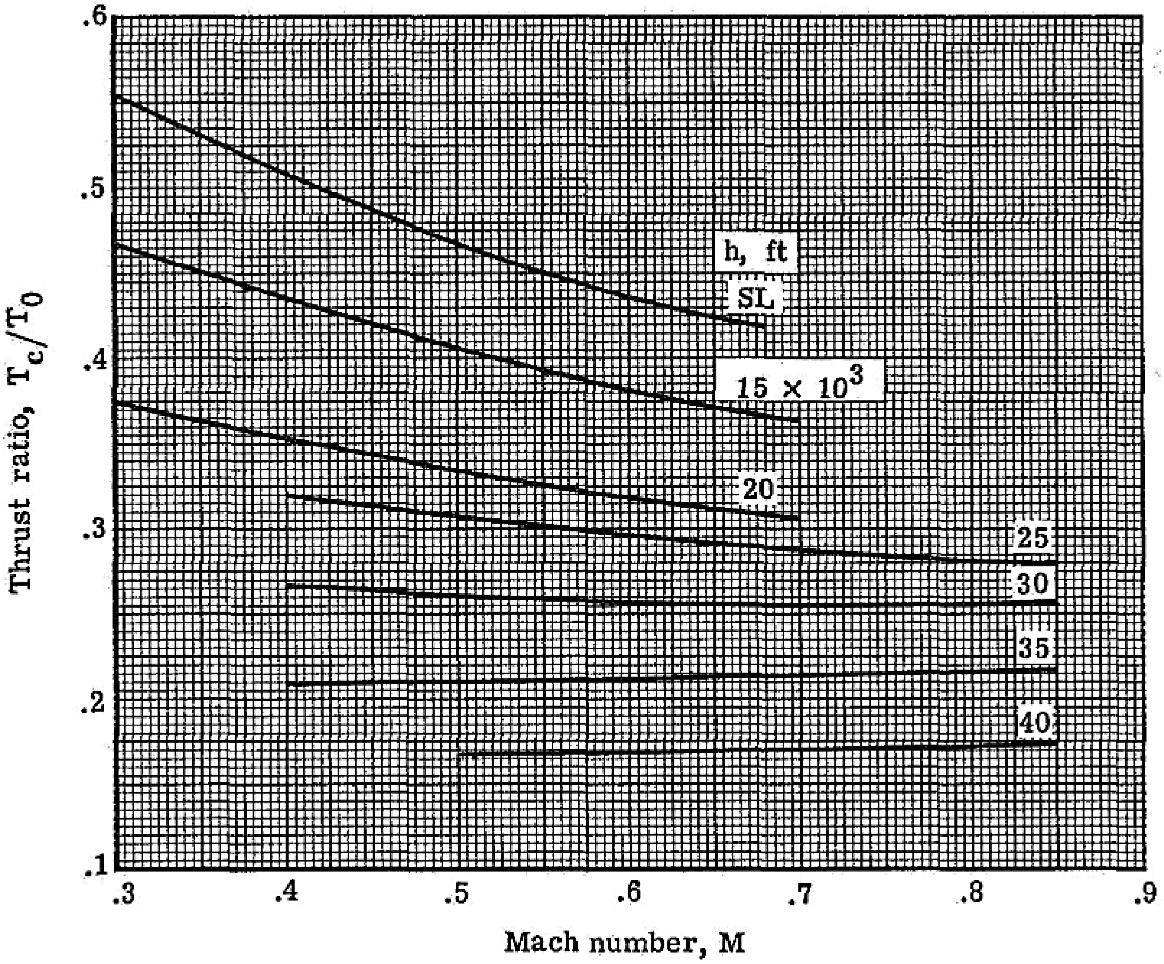


Figure 2.6 Thrust relationships (**Loftin 1980**)

2.5.2 Wing Loading

The wing loading during cruise can be easely determined since the lift equals the weight. The following formula is obtained:

$$\frac{m_{MTO}}{S_W} = \frac{C_L \cdot M^2}{g} \cdot \frac{\gamma}{2} \cdot p(h) \tag{2.17}$$

$$\gamma = 1{,}4$$

The wing loading is depending on the pressure and so it is also depending on the altitude. To predict the wing loading, it is important to know how the pressure develops with the altitude. Therefore we start in the troposphere; the bottom part of the earth's atmosphere which goes from an altitude 0 km to 11 km. In this region, the temperature decreases with 6,5 K/km. The region above the troposphere is called the stratosphere and ends at an altitude of 47 km. For commercial jet transport, only the lower part of the stratosphere will be considered (altitude up to 20 km). In this region of the stratosphere, the temperature and thus the speed of sound remains constant. The pressure for both regions are calculated as follows:

$$p(h) = p_0 \cdot \left(1 - \frac{L \cdot h}{T_0}\right)^{\frac{g \cdot M}{R \cdot L}} = p_0 \cdot \left(1 - \frac{h}{0{,}02256\,km}\right)^{5{,}258} \tag{2.18}$$

$$p_0 = 101325 \text{ Pa}$$
$$L = 0{,}0065 \text{ K/m}$$
$$T_0 = 288{,}15 \text{ K}$$
$$g = 9{,}81 \text{ N/kg}$$
$$M = 0{,}0289644 \text{ kg/mol}$$
$$R = 8{,}31447 \frac{\text{J}}{\text{mol} \cdot \text{K}}$$

Figure 2.7 shows how the pressure changes in function of the altitude. Since the other values in formula (2.17) are fixed, the wing loading will know the same progress in function of the altitude.

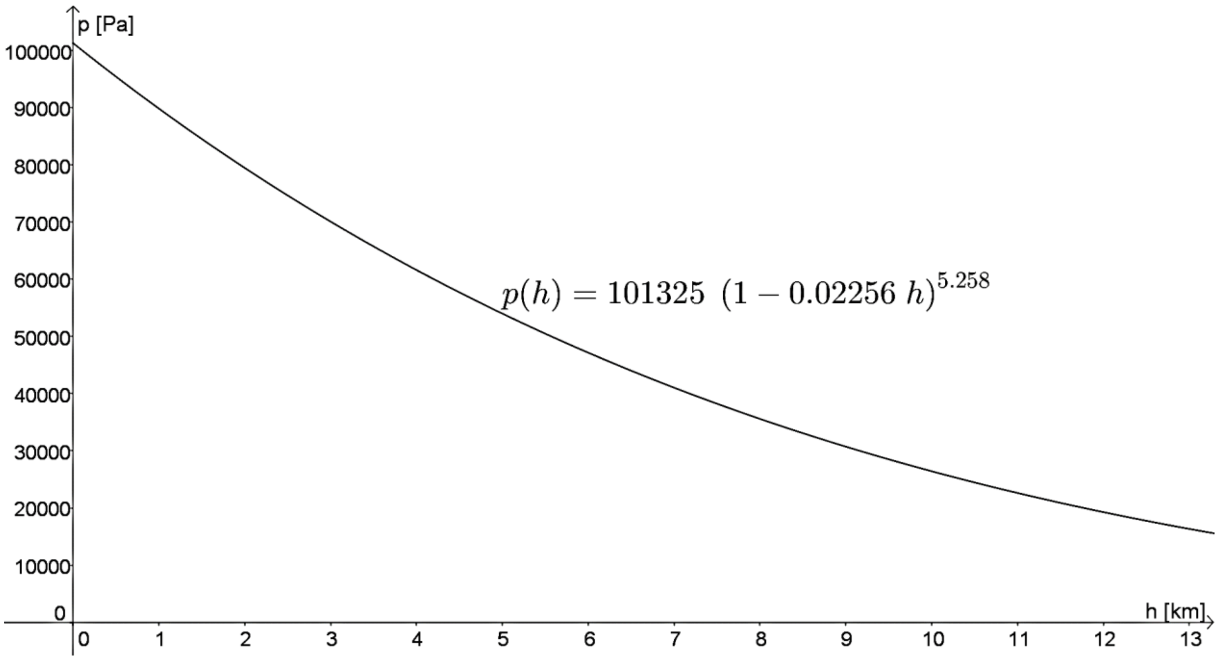


Figure 2.7 Pressure as a function of the altitude

To plot the thrust-to-weight ratio in function of the wing loading for cruise, they must be related to each other. This relation is made by combining the formulas (2.16) and (2.17), using the altitude as the coupling parameter.

2.6 Matching Chart and Design Point

The matching chart is being drawn up by plotting the thrust-to-weight ratio and corresponding wing loading for each segment of the preliminary sizing (landing, take-off, second segment, missed approach and cruise). Figure 2.8 shows a representation from a common matching chart.

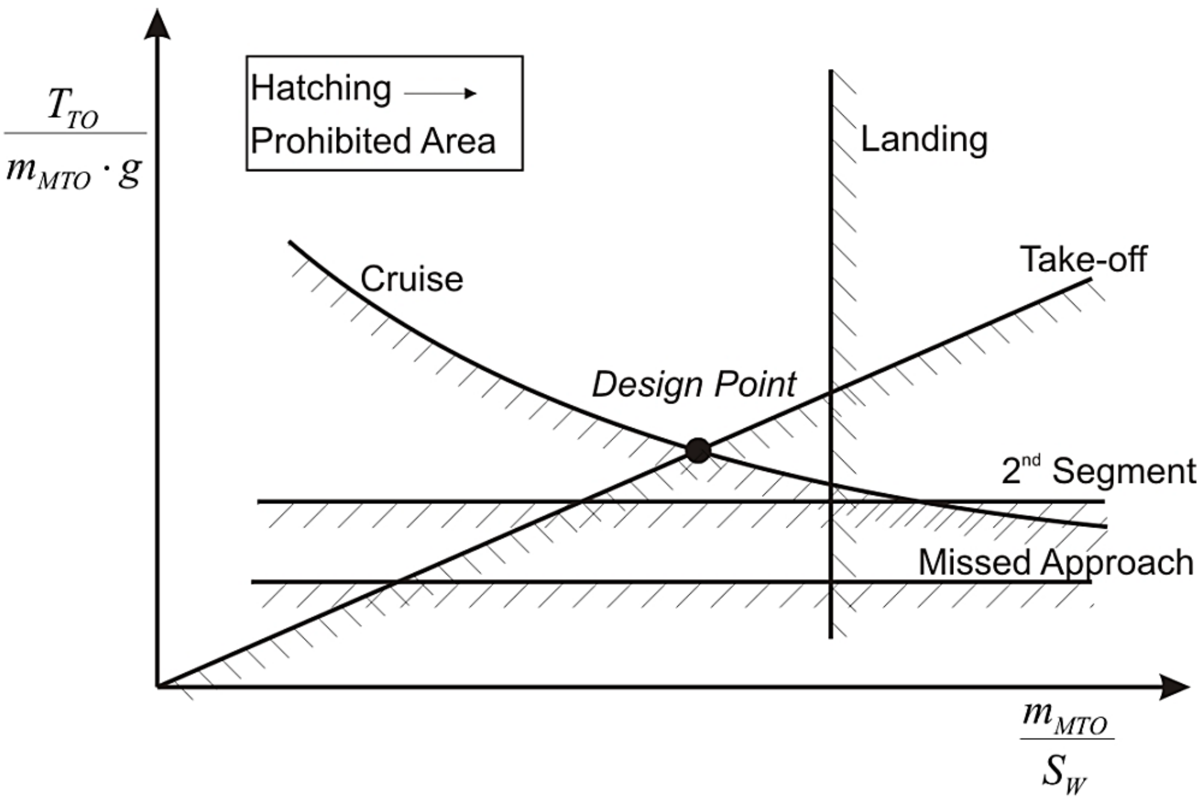


Figure 2.8 Matching chart (**Scholz 2015**)

When designing an aeroplane, the wing loading must be as big as possible. Because we want a lightweight design, the wing can be made smaller when the wing loading is larger. Another important aspect is the thrust-to-weight ratio which has to be as small as possible. A small ratio means smaller engines and thus less operating empty mass. Conclusion; the optimum design point is the point on the matching chart which is at the far right bottom of the graph, while maintaining enough thrust-to-weight ratio for every manoeuvre and not exceeding the wing loading. The closer the lines for every section are to the design point, the better the design.

2.7 Estimation of the Aircraft Design Parameters

The main goal of preliminary sizing is to make an estimation of the aircraft design parameters by knowing only the performance criteria. To do so, it is important that the design point is known from the matching chart. Once this is known, several calculations are made to obtain every design parameter. Starting by calculating the cruise speed, using the thrust-to-weight ratio to calculate the T_{CR}/T_{TO} ratio with the formula (2.15). With (2.16) the altitude is determined. Knowing the altitude, it is possible to calculate the temperature and with that, the speed of sound. In the troposphere, the temperature decreases with 6,5 K for each kilometer increased in altitude (2.19). If the temperature exceeds 216,65 K, the cruise altitude is still in the troposphere. When it is smaller than this value, the cruise altitude is in the stratosphere thus the temperature is constant. Formula (2.20) shows the relation between the temperature and the speed of sound. Since the cruise Mach number is a cruise performance requirement, the cruise speed is calculated as follows (2.21).

$$\begin{cases} if\ T > 216{,}65\ K, \quad T = 288{,}15 - 6{,}5 \cdot h_{CR} \\ \quad else\ T = 216{,}65 \end{cases} \tag{2.19}$$

$$a = 20{,}05 \cdot \sqrt{T} \tag{2.20}$$

$$V_{CR} = M_{CR} \cdot a \tag{2.21}$$

The next parameter that has to be calculated is the relative fuel mass. This is the percentage of the maximum take-off weight that will be taken in fuel mass. Therefore it is important to know how much fuel is used for each segment of the flight mission (Figure 2.9). The fuel fraction represents the amount of total fuel that is used for a segment in percentage (Table 2.1).

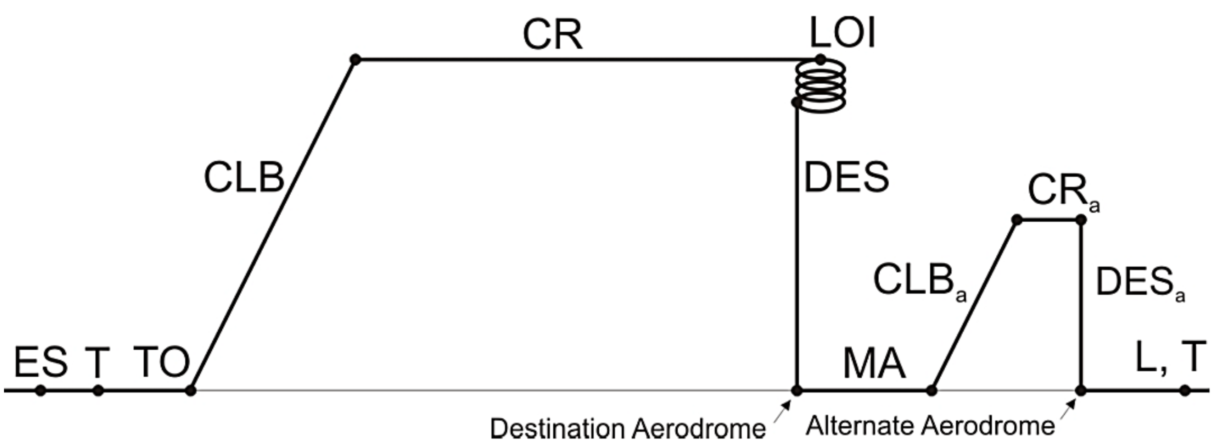


Figure 2.9 Typical flight phases of a civil transport flight mission (**Scholz 2015**)

Table 2.1 Generic mission segment mass fractions (based on **Roskam 1989**)

Mission	Business jet	Jet transport
Engine start	0,990	0,990
Taxi	0,995	0,990
Take-off	0,995	0,995
Climb	0,980	0,980
Descent	0,990	0,990
Landing	0,992	0,992

Notice that Table 2.1 does not contain the fuel fraction for cruise and loiter. This is because it depends on the design and the purpose of the aeroplane. The mission fuel fraction for cruise and loiter are calculated by using formula (2.22) where B is the Breguet factor (2.23) (2.24).

$$M_{ff} = e^{-R/B} \tag{2.22}$$

$$B_s = \frac{E \cdot V_{CR}}{SFC \cdot g} \tag{2.23}$$

$$B_t = \frac{E}{SFC \cdot g} \tag{2.24}$$

To find the fuel fraction, the range has to be known for cruise and loiter. In case of cruise, the design range is known because it is a cruise performance requirement. To make sure the aeroplane will reach its destination, an extra 200 NM are added. For international flights the range increases even more with 5% from the design range. This is imposed by FAR (Part 121). Now the fuel fraction is calculated (2.22) seperatly for the design range and for the extra flight distance where the Breguet factor is expressed in meters (2.23).

The fuel fraction for loiter is similar. The only difference is that the Breguet factor is expressed in seconds because, according to FAR Part 121, the loiter time for domestic flights is 2700 seconds and for international flights it is 1800 seconds.

Eventually, the relative fuel mass can be determined (2.26) using the mission fuel fraction which is calculated in (2.25). Notice that the fuel fraction for engine start and taxi is not implemented. This is because the relative fuel mass belongs to the mission part where the airplane is airborne. The relative amount of fuel needed in reality is given by (2.27).

$$\begin{aligned} M_{ff} = {} & M_{ff,TO} \cdot M_{ff,CLB} \cdot M_{ff,CR} \cdot M_{ff,DES} \cdot M_{ff,L} \cdot \left(M_{ff,RES} \cdot M_{ff,CLB}\right) \\ & \cdot \left(M_{ff,loiter} \cdot M_{ff,DES}\right) \end{aligned} \tag{2.25}$$

$$\frac{m_F}{m_{MTO}} = 1 - M_{ff} \tag{2.26}$$

$$\frac{m_{F,needed}}{m_{MTO}} = 1 - M_{ff} \cdot M_{ff,engine\ start} \cdot M_{ff,taxi} \tag{2.27}$$

The payload is a cruise performance requirement. The relative operating empty mass can be estimated with equation (2.28). And knowing the relative fuel mass, it is possible to calculate the maximum take-off mass (2.29), the operating empty mass and the fuel mass.

$$\frac{m_{OE}}{m_{MTO}} = 0{,}23 + 1{,}04 \cdot \frac{T_{TO}}{m_{MTO} \cdot g} \tag{2.28}$$

$$m_{MTO} = \frac{m_{PL}}{1 - \frac{m_F}{m_{MTO}} - \frac{m_{OE}}{m_{MTO}}} \tag{2.29}$$

The last design parameters are calculated using the wing loading and the thrust-to-weight ratio with the following formulas:

$$S_W = m_{MTO} \cdot \frac{S_W}{m_{MTO}} \tag{2.30}$$

$$T_{TO} = m_{MTO} \cdot g \cdot \frac{T_{TO}}{m_{MTO} \cdot g} \tag{2.31}$$

3 Reverse Engineering

3.1 Definition and Use

Reverse engineering is used in a wide range of application areas. Therefore it is known under many different definitions, depending on the context. Reverse engineering can be applied in cases as:

- Exploiting software
- Creating a CAD model of a product
- Providing technological documentation (regarding design, development or manufacturing processes from parts or assemblies)
- As initial process for a redesign
- To study the working and effect of a product or system

In general, reverse engineering can be devided in two main groups; software related – and mechanical related reverse engineering. Since this study is focussing on the design of an aeroplane, only the mechanical related reverse engineering will be explained.

In terms of mechanical reverse engineering, it is a systematical approach to build up a duplication of an already excisting product or system, while there are no documents or drawings available. With this process, one is able to discover the technological principles and ideas behind a fully functional excisting device.

The first case where an interesting use of reverse engineering could be made, is in case of failure or obsolescence from devices, parts and systems in the industry (e.g. fan blades). This means that the part has to be changed. As long as there is a supply or distributor for these parts, there is no problem. Unfortunately, it frequently happens that the product is no longer available. And with some bad luck, there are no specifications of the existing product. In this case, the owner of the industrial site has three options.

The first option is to find another suitable part which replaces the previous one. This solution is cheap, but again, you are depending on a distributor.

The second option is to re-engineer the entire system. This is expensive and takes a lot of time. In this case, much effort is used in that is embedding relation to the size of the actual problem.

The last option is to reverse engineer the broken part to regain the detailed specifications of the part. This solution is also expensive, but you own the specifications of the part. If by doing so, the advantage can be increased when one is able to produce the part by its own means. The owner now is fully independent of a distributor.

An additional reason to use reverse engineering is the fact that companies have to be competitive at all times. This is only possible when they are able to come up with new, better or upgraded products. In case they want to improve an excisting product of their own, they can reverse engineer the existing product and improve it. To improve the product, companies need to know what the requirements for the customer are. During the reverse engineering, they connect the requirements to the performances of the product. This way, they can come up with innovative solutions and get a better market position.

In the latter case, where reverse engineering is used, the functioning of a product or system has to be discovered. It starts with a finished, working product, which will be disassembled and examined into the smallest detail. This is ofton the motive for many manufacturers which want to study the products of their competitors and learn from the developments made by them. Another reason could be out of interest or research.

In general, reverse engineering is a process, used to understand the working of a product and to analyze its design. It is applied when the technical data got lost, don't excist, or is the property of another manufacturer (who doesn't excist anymore, has shut down the production or wants to protect the specifications of their product). Reverse engineering makes it possible to provide documentation of the product which contains the specifications, mechanisms, materials, manufacturability and assembly details.

3.2 Permissibility to Perform Reverse Engineering

In this case, the aim is to dissect a designed aeroplane using reverse engineering. By doing this, specific parameters are revealed which, in most designs, are concealed by the designing company. Since this case is a study, guided by the Hochschule für Angewandte Wissenschaften Hamburg, an exception on intellectual ownership is applicable. This means that it is not necessary to ask the owners of the copyright for permission to reproduce or publicly share the protected information.

3.3 Methodology of Reverse Engineering

This case is about the reverse engineering of aeroplanes. Since the study is theoretical, it is not necessary to do a teardown of an aeroplane, which is also possible. This means that the usual reverse engineering method for mechanical devices can not be applied. Instead, a theoretically reverse engineering method will be explained which still belongs to the mechanical related reverse engineering, according to (**Otto 2001**).

The theoretically reverse engineering method consists out of two steps. The first step is the prescreening of the product and the build-up of a black-box. Therefore, information is gathered and the inputs and outputs of the product are written down. The first step is essential to be able to perform the second step in a structured way. The second step is the functional analysis. Here, the mutual relations of the inputs and outputs are studied and determined.

3.4 STEP 1: Prescreening and Black-box

When a product needs to be reverse engineered, the first step of the process is the prescreening of the product. This includes investigation, prediction and hypothesis. It is important to put effort in the research of the product to obtain as much available information as possible. Unfortunately, not all the specifications are available since reverse engineering is necessary. After this, the black-box model is build-up.

Every product is developed for a reason. This reason could be, for example, a customer who needs a certain machine which has to meet the customers' expectations. To develop a black-box, these expectations are described as intendend functions of a product. The purpose of the black-box model is to transform these intendend functions into outputs and make a causal link with inputs. This means that the black-box has a hypothetical form. Eventually, the black-box model gives a graphical overview of the inputs and outputs of the product without knowing the mutual relation.
Figure 3.1 gives an example of a black-box build-up for a simple rotary tool.

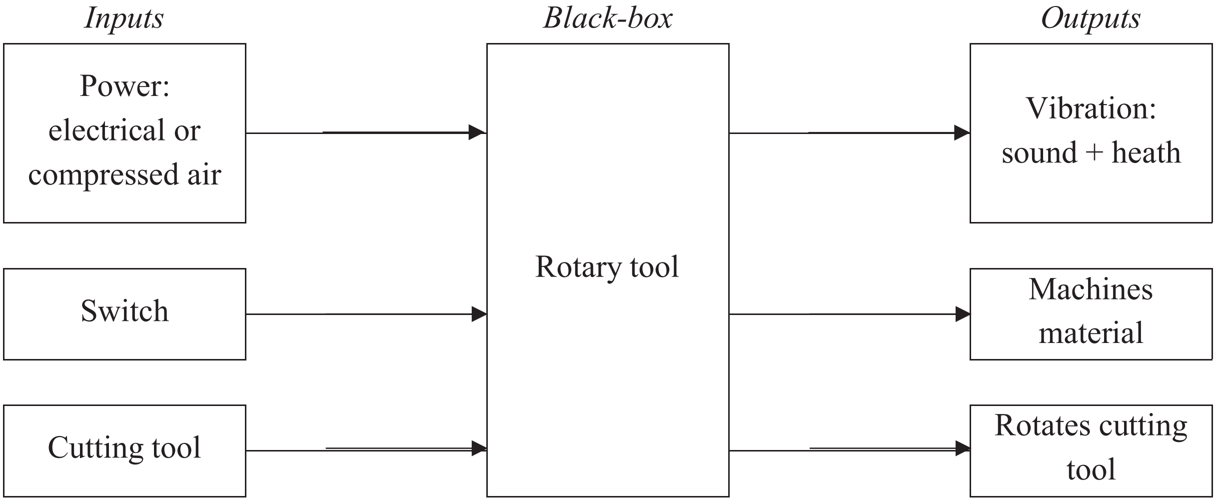


Figure 3.1 Black-box for a rotary tool

3.5 STEP 2: Functional Analysis

The second step is called 'the functional analysis'. This contains the set up of relations between the inputs and outputs of a product while aiming towards the expectations of the customer. This is done by creating functions. A function describes the mutual relation between the inputs and outputs of the product. To create functions, it is important to know the tasks of the product. For each task, there is a function which links it to an input. When there are more inputs that are linked in a different way to the output, it is possible to make subtasks that are represented by subfunctions. The black-box is in this phase of the reverse engineering usefull to find the functions and subfunctions. The functions are determined by input-output-relations or constraints.

The intput-output-relations can be found with formulas. Therefore, the relations between the input and output must be discovered. It is important to know which parameters effect the output and in which size the output gets changed by it. This can be done theoretically or empirically.

For the theoretical approach, the relations between input and output are already studied and are available in books. Another way is to use different formulas and derive an own formula that describes the relation between the inputs and outputs.

Constraints are the result of the expectations of the customer so it is a criterion that must be fulfilled by the product. By using constraints in formulas, the amount of variables reduces, resulting in a smaller range of values for the output.

In the end, the black-box is expanded and becomes a white-box. A white-box knows even the smallest relations between inputs and outputs are known. The function can consist of inputs, which are parameters, constraints or variables. In case of variables, the value is the one which optimizes for the output. When there are too many variables, the amount has to be decreased by adding more relations with more inputs.

3.6 Summary of the Reverse Engineering Process

The reverse engineering starts with the research of the product. The next step is the build-up of the black box which consists out of the inputs and outputs of the product without knowing the mutual relation. To find the internal relations between the inputs and outputs, a function analysis is performed. This results in functions that are determined by input-output-relations and contraints. Eventually the black-box is transformed into a white-box.

Conclusion, to perform a reverse engineering process, a knowledge of several engineering areas is required. The entire process starts with the understanding of the product, how the separate parts work together. What is their function? What is their mutual interaction? Thereafter the reverse process starts, which requires skills in problem solving. In the end, the product is theoretically reverse engineered and the inputs and outputs are determined in a way that the product can satisfy the requirements of the customer.

4 Reverse Engineering in Aircraft Preliminary Sizing

Common specifications for commercial aeroplanes are easily to find, but there are a few exceptions. These exceptions are called ‘the companies’ secrets’. These parameters are not released by the designcompany because that way everybody could produce duplicates of the design and all the investments of research, work and money could be abused by third parties. But there is a way to find these parameters. By uniting the knowledge of preliminary sizing and reverse engineering, a good approximation of these parameters can be made. These parameters are the maximum lift coefficient for landing and take-off, the maximum aerodynamic efficiency and the specific fuel consumption.

The main goal of this thesis, is to provide a program which makes it possible to gain the classified information in a quick and accurate manner. Because it has to be quick and user friendly, it is important that the number of inputs is limited. Therefore, the calculations are only based on the preliminary sizing and not a conceptual or final design approach.

4.1 STEP 1: Prescreening and Black-box

As mensioned before, the theoretical reverse engineering starts with the prescreening of the product. Therefore, a product must be chosen, in this case a certain airplane is selected. To determine the reverse engineering parameters from the selected airplane, it is important that the common specifications of the concerned aeroplane are known. Therefore it is prescreened by doing research on information about the airplane specifications. To perform a succesfull reverse engineering, it is important that the following specifications for jet powered aeroplanes are known from the prescreening:

Table 4.1 Necessary specifications for jet powered aeroplanes

Quantity name	Symbol	SI Unit
Landing field length	s_{LFL}	m
Relative air density (landing)	σ	-
Take-off field length	s_{TOFL}	m
Relative air density (take-off)	σ	-
Design range	R	m
Cruise Mach number	M_{CR}	-
Wing area	S_W	m²
Span	b	m
Relative landing mass	m_{ML}/m_{MTO}	-
Maximum take-off mass	m_{MTO}	kg
Operating empty mass	m_{OE}	kg
Maximum payload mass	m_{PL}	kg
Wing loading	m_{MTO}/S_W	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	-
Bypass ratio	μ	-
(Available fuel volume	$V_{fuel,available}$	m³)
Cruise speed	V_{CR}	m/s
Cruise altitude	h_{CR}	m

These values can be found on the site of the aeroplane manufacturer. For example, if a pre-screening has to be made for an A340, than the information can be found on the site of Airbus with the keywords 'airport planner'. If this source doesn't provide enough information for the prescreening, other sites or books might do. Some values are not always available, paragraph 5.9 describes tips and tricks to find or estimate them. Next to these values, other parameters are necessary. These are known constants or constants based on experience. An overview of these parameters is shown in Table 4.2:

Table 4.2 Constant parameters

Quantity name	Symbol	Value	SI Unit
Gravitational acceleration	g	9,81	m/s²
Ratio of specific heats, air	γ	1,4	-
Standard static pressure at sea-level	p_0	101325	N/m²
Fuel density	ρ_{fuel}	800	kg/m³
Oswald efficiency factor, clean	e	0,85	-

Once the prescreening is done, the next step is to build-up the black-box. In this case, it is possible to base the black-box on Figure 2.1. The outputs are the reverse engineering results; maximum lift coefficient for landing and take-off, maximum aerodynamic efficiency and the specific fuel consumption. The inputs are the aeroplanes specifications shown in Table 4.1. To make things easier, subfunctions are implemented on; landing, take-off and cruise. The sub-function cruise consists out of two additional subfunctions, because it contains relations for two outputs that are determined a different way. As a result, Figure 4.1 shows the final black-box for the reverse engineering process for jet powered aeroplanes.

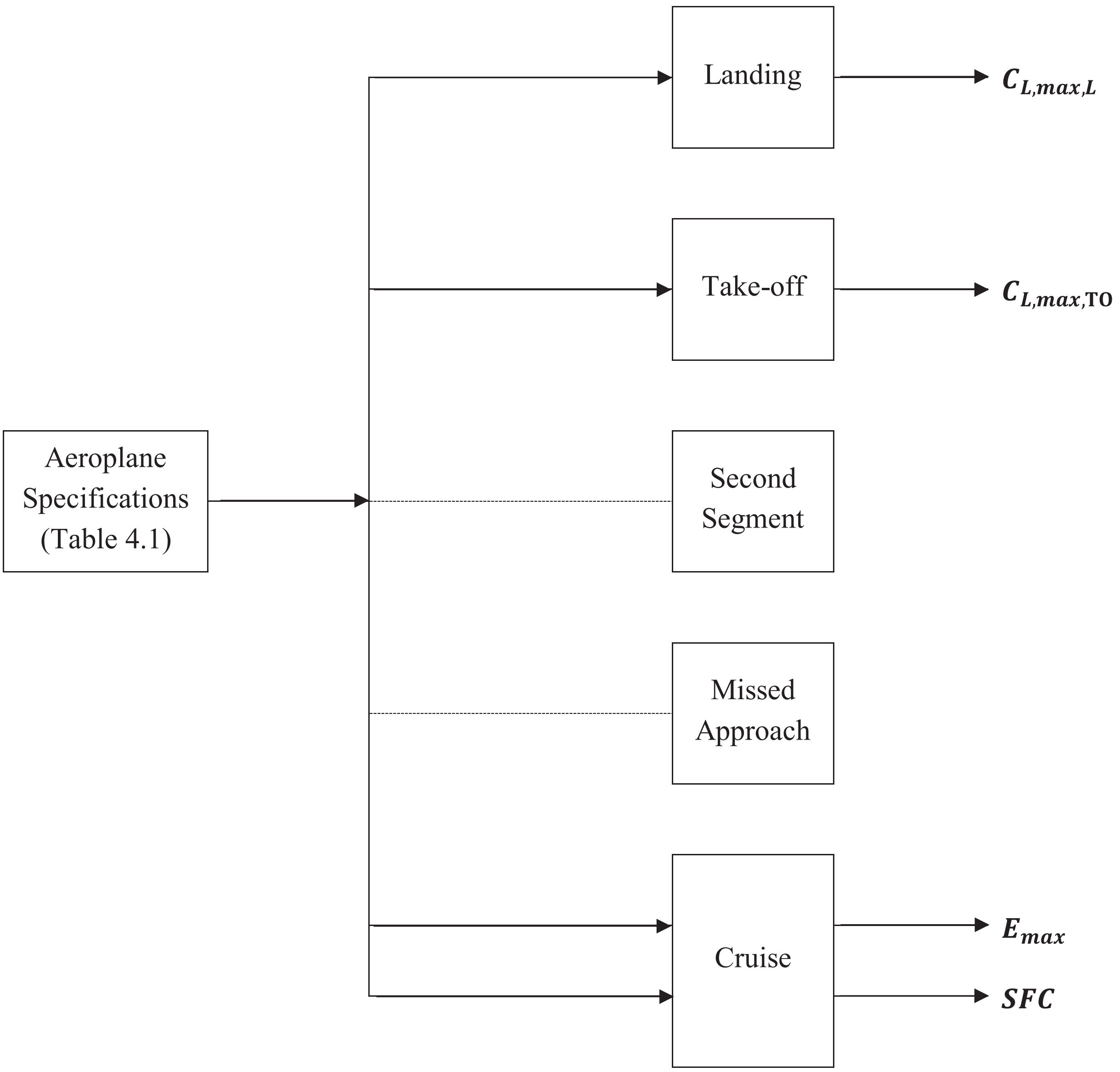


Figure 4.1 Black-box for jet powered aircraft

4.2 STEP 2: Functional Analysis

4.2.1 Maximum Lift Coefficient for Landing and Take-off

Since the design itself is not involved anymore, but only the quest for the reverse engineering values, the wing loading and thrust to weight ratio are known. From the matching chart of aircraft preliminary sizing (Figure 2.8), we know that landing is represented by a vertical line. This means that the landing is a constraint for only the wing loading while it is also independent from the thrust-to-weight ratio. Thus, the maximum lift coefficient for landing can be calculated with an adaption of formula (2.2) which becomes equation (4.1):

$$C_{L,max,L} = \frac{\frac{m_{MTO}}{S_W} \cdot \frac{m_{ML}}{m_{MTO}}}{k_L \cdot \sigma \cdot s_{LFL}} \tag{4.1}$$

$$k_L = 0{,}107\ kg/m^3$$

To calculate the maximum lift coefficient for landing, it is nessecary that the value of the wing loading, the relative maximum landing mass, the weather conditions and the landing field length are known. If the landing field length is not known, but the approach speed is, you can determine this length with formula (2.1). If this value is also unknown, paragraph 5.9 gives some tips and tricks to find the reverse engineering values without knowing the exact landing field length or approach speed.

The same reasoning applies for the maximum lift coefficient for take-off. The matching chart shows a positive linear relation between the wing loading and thrust-to-weight ratio for take-off. We assume that the designers aimed for an optimum design point, where all lines should cross eachother in one point. This means that the known thrust-to-weight ratio should occur at the known wing loading for take-off. By adapting formula (2.3) we get (4.2) which makes it possible to calculate the maximum lift coefficient for take-off.

$$C_{L,max,TO} = \frac{k_{TO}}{\sigma \cdot s_{TOFL}} \cdot \frac{m_{MTO}/S_W}{T_{TO}/m_{TO} \cdot g} \tag{4.2}$$

$$k_{TO} = 2{,}34\ m^3/kg$$

If the wing loading, the weather conditions, the take-off field length and the thrust-to-weight ratio are known, it is possible to calculate the maximum lift coefficient for take-off. If there are difficulties to find the excact value for the take-off field length, paragraph 5.9 gives a possible solution for this problem.

The black-box for the maximum lift coefficient for landing and take-off, turns into a white-box shown in Figure 4.2:

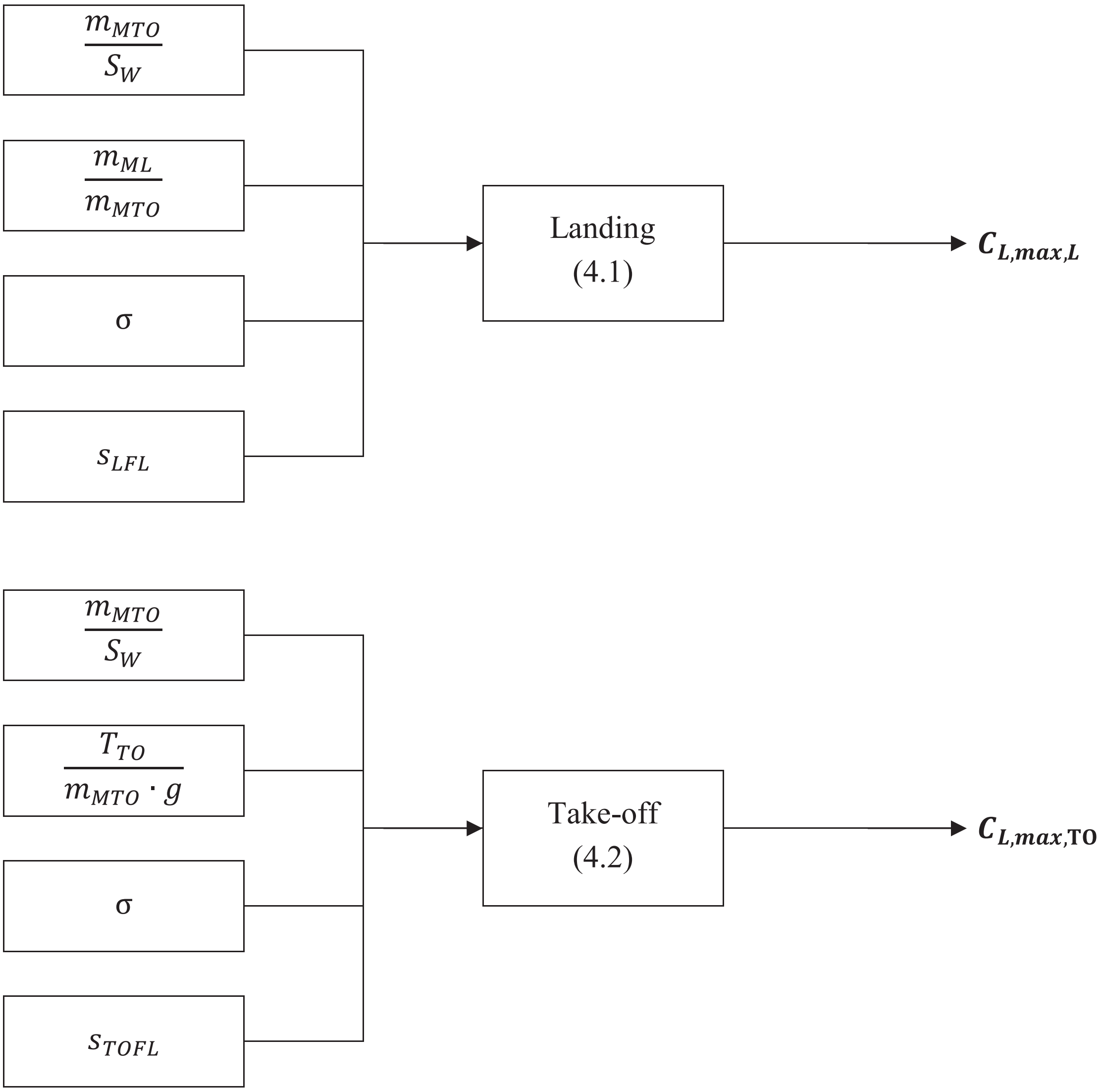


Figure 4.2 White-box for maximum lift coefficient

4.2.2 Maximum Aerodynamic Efficiency

The most complex calculation of all the reverse engineering values, is the one to find out the maximum aerodynamic efficiency, or aerodynamic efficiency. In the preliminary sizing of an aircraft, E_{max} is determined with formula (2.13). Then several parameters can be calculated because they are depending on the maximum aerodynamic efficiency. But in the reverse direction, those parameters are also unknown, resulting in a lot of unknown parameters and are thus unusable to calculate the maximum aerodynamic efficiency.

To find a formula for the maximum aerodynamic efficiency, we use the cruise section from the preliminary sizing. In this section, the maximum aerodynamic efficiency is used to determine the zero-lift drag $C_{D,0}$ (4.3) and the actual aerodynamic efficiency (2.11) during cruise.

The zero-lift drag is used to calculate the minimum drag lift coefficient (4.4). With this parameter and assuming that cruise speed to minimum drag speed ratio V/V_{md} is known, it is possible to determine the lift coefficient during cruise (4.5). This lift coefficient is needed to determine the wing loading (4.6).

$$C_{D,0} = \frac{\pi \cdot A \ e}{4 \cdot E_{max}^2} \tag{4.3}$$

$$C_{L,md} = \sqrt{C_{D,0} \cdot \pi \cdot A \ e} \tag{4.4}$$

$$C_L = \frac{C_{L,md}}{\left(\frac{V}{V_{md}}\right)^2} \tag{4.5}$$

$$\frac{m_{MTO}}{S_W} = \frac{C_L \cdot M^2}{g} \cdot \frac{\gamma}{2} \cdot p_0 \cdot (1 - 0{,}02256 \cdot h)^{5{,}258} \tag{4.6}$$

The aerodynamic efficiency is used to calculate the necessary thrust-to-weight ratio (2.15) where T_{CR}/T_{TO} is a function of the altitude (2.16). In the preliminary sizing, the thrust-to-weight ratio and wing loading are joined by using the altitude. For reverse engineering, this means that the additional difficulty is to match the thrust-to-weight ratio and the wing loading in a correct way. Therefore it is interesting if the formulas (2.16) and (4.6) are adapted in a way that the altitude is the function and that both are expressed in the same unit, kilometers:

$$h = \frac{1}{0{,}02256} \left[1 - \left(\frac{2 \cdot g \cdot \frac{m_{MTO}}{S_W}}{C_L \cdot M^2 \cdot \gamma \cdot p_0} \right)^{\frac{1}{5{,}258}} \right] \tag{4.7}$$

$$h = \frac{\frac{T_{CR}}{T_{TO}} + 0{,}0248\mu - 0{,}7125}{0{,}0013\mu - 0{,}0397} \tag{4.8}$$

The maximum aerodynamic efficiency can now be written as an equation with rational exponents. This equation is composed by combining the equations: (2.11), (2.15), (2.16), (4.3), (4.4), (4.5) and (4.7). Solving equation (4.9) gives a solution for the maximum aerodynamic efficiency.

In chapter 5 The Tool, it will be clear that Excel has problems with this equation. To bypass this problem, the Newton-Raphson method is integrated into the Excel. To use this method, the derivative of equation (4.9) is needed (4.10).

$$\frac{2\cdot\frac{T_{TO}}{m_{MTO}\cdot g}}{\frac{1}{\left(\frac{V}{V_{md}}\right)^2}+\left(\frac{V}{V_{md}}\right)^2}\cdot\left[E_{max}^{1,19}\cdot(0{,}0576\mu-1.76)\cdot\left(\frac{4\cdot g\cdot\frac{m_{MTO}}{S_W}\cdot\left(\frac{V}{V_{md}}\right)^2}{\pi\cdot\ A\ \ e\ \ M^2\cdot\gamma\cdot\ p_0}\right)^{\frac{1}{5.258}}\right]$$

$$\frac{2\cdot\frac{T_{TO}}{m_{MTO}\cdot g}}{\frac{1}{\left(\frac{V}{V_{md}}\right)^2}+\left(\frac{V}{V_{md}}\right)^2}\cdot(-E_{max})\cdot(0{,}0328\mu-1{,}05)\ +1=0 \tag{4.9}$$

$$1{,}19\cdot E_{max}^{\frac{1}{5.258}}\cdot(0{,}0576\mu-1.76)\cdot\left(\frac{4\cdot g\cdot\frac{m_{MTO}}{S_W}\cdot\left(\frac{V}{V_{md}}\right)^2}{\pi\cdot\ A\ \ e\ \ M^2\cdot\gamma\cdot\ p_0}\right)^{\frac{1}{5.258}}$$

$$-(0{,}0328\mu-1{,}05)=0 \tag{4.10}$$

Previous equations make it possible to calculate the maximum aerodynamic efficiency while the relation between the wing loading and thrust-to-weight ratio in cruise is remained correctly. The equation makes only use of values which are known by the specifics of the aeroplane; thrust-to-weight ratio, wing loading, aspect ratio A, Oswald's span efficiency factor e, Mach number M, ratio of specific heats for air γ and the standard static pressure at sea-level p_0. The cruise speed to minimum drag speed ratio V/V_{md} depends on the design, wether one wants to fly at maximum aerodynamic efficiency or at maximum range. The bypass ratio of the engines μ is known according to the engine specifications. Eventually only the maximum aerodynamic efficiency remains as the only unknown parameter in the function. This can be represented with a white-box, shown on Figure 4.3.

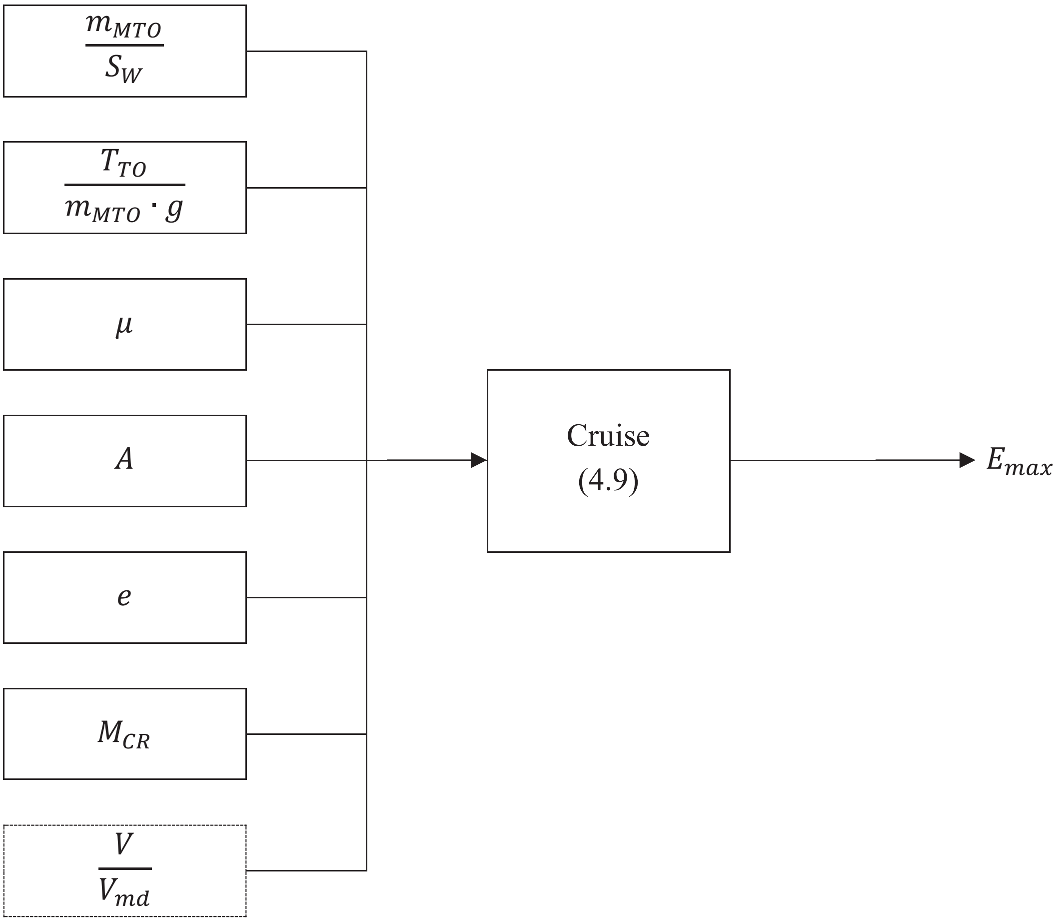


Figure 4.3 White-box for maximum aerodynamic efficiency

4.2.3 Specific Fuel Consumption

The specifications of the aeroplane contain the operating empty mass and the payload mass. Using these parameters, the relative fuel mass can be calculated using equation (2.29). When this is known, the specific fuel consumption can be calculated using the following equation.

$$SFC = -\frac{E \cdot \ ln\left(\frac{\frac{m_{PL}}{m_{MTO}} + \frac{m_{OE}}{m_{MTO}}}{M_{ff,TO} \cdot M_{ff,CLB}^2 \cdot M_{ff,DES}^2 \cdot M_{ff,L}}\right)}{g \cdot \left(\frac{R+\ s_{RES}}{V_{CR}} + t_{loiter}\right)} \tag{4.11}$$

If the maximum range is assumed, the available fuel volume (fuel capacity) can also be used. To calculate the fuel consumption, this is the only parameter that is nessecary, other parameters are already determined in previous calculations. The idea is; by knowing the maximum amount of fuel, the minimum mission fuel fraction can be calculated, relying on equation (2.27). This mission fuel fraction will be discomposed until there will be one equation for the specific fuel consumption. So first thing is to determine the mission fuel fraction by knowing the maximum amount of fuel.

$$m_{fuel,available} = V_{fuel,available} \cdot \rho_{fuel} \qquad (4.12)$$
$$\rho_{fuel} = 800 \text{ kg/m}^3$$

$$M_{ff} \cdot M_{ff,engine\,start} \cdot M_{ff,taxi} = 1 - \frac{m_{fuel,available}}{m_{MTO}} \qquad (4.13)$$

At this point, the left part of the equation is known. Now this part wil be decomposed until only the specific fuel consumption remains as an unknown parameter. Equation (2.25) is used to replace the mission fuel fraction.

$$M_{ff} = M_{ff,TO} \cdot M_{ff,CLB} \cdot M_{ff,CR} \cdot M_{ff,DES} \cdot M_{ff,L} \cdot \left(M_{ff,RES} \cdot M_{ff,CLB}\right) \cdot \left(M_{ff,loiter} \cdot M_{ff,DES}\right) \qquad (4.14)$$

Only the fuel fraction of cruise, reserves and loiter are depending on the specific fuel consumption. Based on the Breguet formulas (2.22), (2.23) and (2.24), from preliminary sizing in paragraph 2.7, the following equations are drawn up:

$$M_{ff,CR} = e^{-\frac{\frac{R}{E \cdot V_{CR}}}{SFC \cdot g}} \qquad (4.15)$$

$$M_{ff,RES} = e^{-\frac{\frac{s_{RES}}{E \cdot V_{CR}}}{SFC \cdot g}} \qquad (4.16)$$

$$M_{ff,loiter} = e^{-\frac{\frac{t_{loiter}}{E}}{SFC \cdot g}} \qquad (4.17)$$

By implementing these three formulas in equation (4.14) and combining this with (4.13), the specific fuel consumption becomes the only unknown parameter in the equation:

$$SFC = -\frac{E \cdot \ ln\left(\frac{1 - \frac{V_{fuel,available} \cdot \rho_{fuel}}{m_{MTO}}}{M_{ff,TO} \cdot M_{ff,CLB}^2 \cdot M_{ff,DES}^2 \cdot M_{ff,L} \cdot M_{ff,engine\,start} \cdot M_{ff,taxi}}\right)}{g \cdot \left(\frac{R + s_{RES}}{V_{CR}} + t_{loiter}\right)} \qquad (4.18)$$

As formula (4.18) shows, the specific fuel consumption *SFC* is determined by the amount of fuel that the aeroplane can carry $V_{fuel,available}$, the remaining percentage of fuel after several stages of the mission (fuel fractions M_{ff}), the designed range *R*, the extra distance to alternate s_{RES}, the loiter time t_{loiter} and the cruise speed V_{CR}. The cruise speed can be determined the same way as it has been done for the preliminary sizing. First the altitude is calculated by using the wing loading (2.17) or the thrust-to-weight ratio (2.16). Then equations (2.19), (2.20) and (2.21) are used in order to find the cruise speed.

The mutual relations between the aeroplane specifications and the specific fuel consumption is now known. The white-box is shown in the following figure:

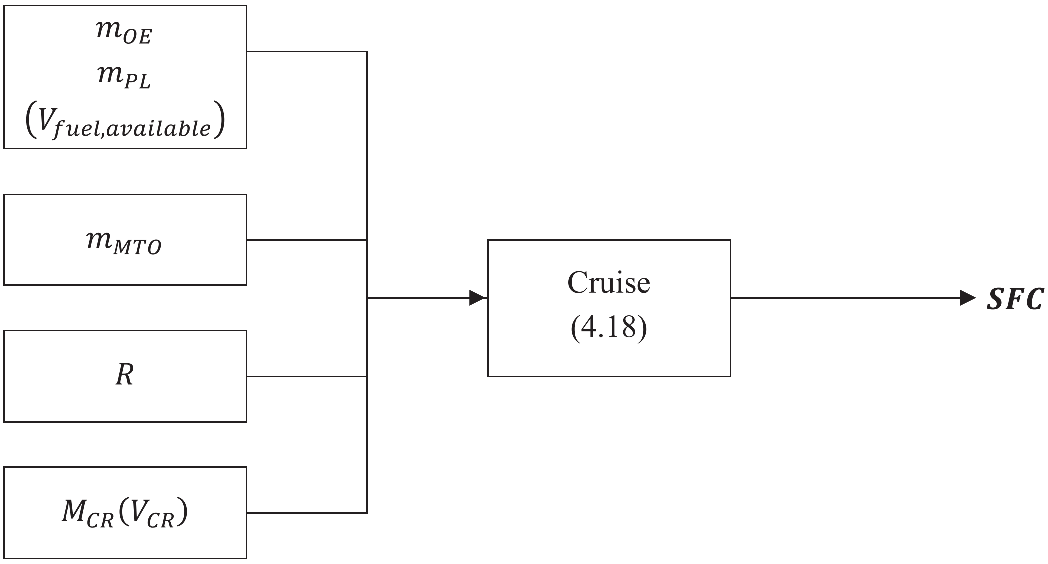


Figure 4.4 White-box for the specific fuel consumption

4.2.4 The Complete White-box

Every function and subfunction is built. The entire black-box can be replaced by a white-box. Figure 4.5 represents the entire reverse engineering process. The inputs are the values found with the prescreening (Table 4.1). The outputs are the reverse engineering values. And the mutual relation is shown by the equations between brackets. The process to reverse engineer an aeroplane consists of out of four subfunctions; landing, take-off and two times climb. Each subfunctions require certain inputs.

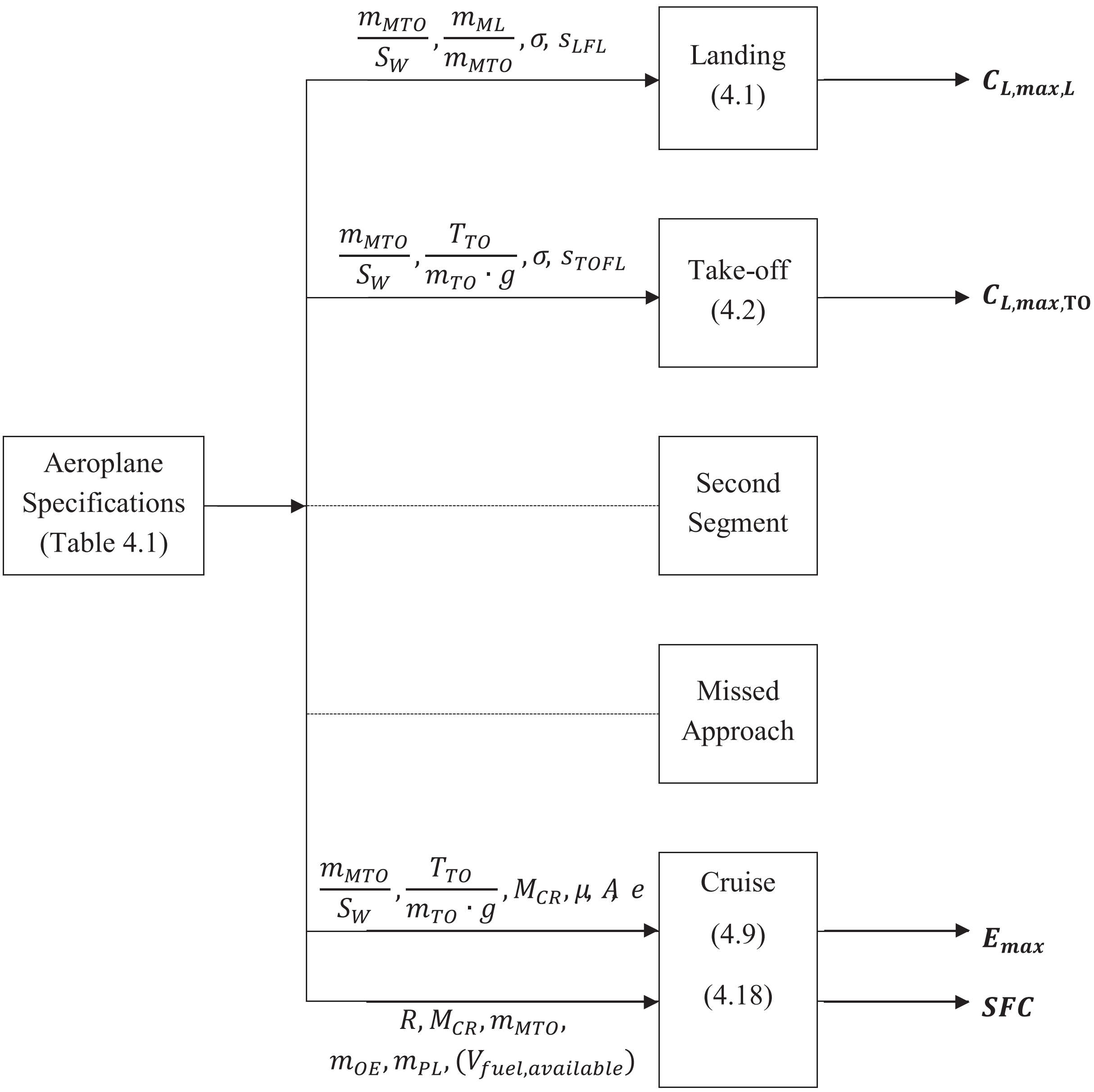


Figure 4.5 Complete white-box for jet powered aeroplanes

4.3 Applying the Reverse Engineering Method

Now that all the reverse engineering values are theoretically discovered, it can be applied to a model. In this chapter, we can conclude that reverse engineering can get very complex in order to keep all the relations between all the parameters and variables. Since this is a complex method and a lot of calculations are requiered, it is interesting to make a tool where only the nessecary values are entered. The tool makes than the calculations and takes the different relations into account. This way, the reverse engineering can be performed with one single push on a button. When a parameter changes, it is not nessecary to redo all the calculations by yourself. Next chapter will describe the build-up and working of the tool.

5 The Tool

The tool used to perform the reverse engineering is an Excel file called 'Reverse Engineering.xlsm'. The file exist out of eight tabs and an extra tab containing instructions. The tabs are: Data, Specs + RE, 1) $C_{L,max}$, 2) E_{max}, 3) SFC, 4) Verification, Matching chart and Matching chart points. In this paragraph, each tab will be discussed in detail.

5.1 Data

This tab contains technical and empirical data that are only required in certain specific cases. The tab is indispensable if there are values unknown or a verification has to be executed. It also contains constants to convert empirical to metric units. Therefore, the Excel file can not operate without consulting this tab.

5.1.1 SKYbrary

SKYbrary is an electronic repository of safety knowledge related to flight operations, air traffic management and aviation safety in general. In this case, it is used as an electronic library to gain information about certain aircraft parameters in different aircraft classes. Using this information, it is possible to get an upper and lower limit for the wing span, take-off field length and the approach speed (and with this, indirectly the landing field length). It is used when one of these values are unknown for a certain aeroplane which has to be reverse engineered.

If this is the case, the aircraft category must be given by the user. Starting with the 'Airplane Design Group' which gives limitations for the wing span and tail height (Table 5.1).

Table 5.1 Aircraft Design Group (**SKYbrary 2017c**)

Aircraft category	Wing span [m]		Tail height [m]	
	LL	UL	LL	UL
I		15		6,1
II	15	24	6,1	9,1
III	24	36	9,1	13,7
IV	36	52	13,7	18,3
V	52	65	18,3	20,1
VI	65	80	20,1	24,4

Secondly, there is the 'ICAO Aerodrome Reference Code' which consists of a numerical class (one to four) and an alphabetical class (A to F). They give limitations for respectively the reference field length and wing span plus outer main gear wheel span (Table 5.2).

Table 5.2 ICAO Aerodrome Reference Code (**SKYbrary 2017d**)

Aircraft category	Aeroplane reference field length [m]	
	LL	UL
1		800
2	800	1200
3	1200	1800
4	1800	

Aircraft category	Wing span [m]		Outer main gear wheel span [m]	
	LL	UL	LL	UL
A		15		4,5
B	15	24	4,5	6
C	24	36	6	9
D	36	52	9	14
E	52	65	9	14
F	65	80	14	16

The third category is the 'Aircraft Approach Category' (Table 5.3) which gives limitations for the approach speed and thus the landing field length (according to equation (2.1)).

Table 5.3 Aircraft Approach Category (**SKYbrary 2017b**)

Aircraft category	Typical aircraft	V_{APP} [kt]	
		LL	UL
A	small single engine		90
B	small multi engine	91	120
C	airline jet	121	140
D	large jet/military jet	141	165
E	special military	166	210

Notice that the classification 'Aircraft Design Group' and 'ICAO Aerodrome Reference Code' both give a value for the wing span. If the user gives an aircraft category for a certain aeroplane and the value for the wing span does not match for both classifications, the user will get a notification 'Conflict between ADG and ICAO'. If this is the case, the user has to verify the inputs for the aircraft category. The operator can assign the matching aircraft category using the drop down menus for every class (Figure 5.1).

AIRCRAFT	1. ADG	2. ICAO	3. AAC
B747-400	V	4 E	D

	Specification limits			
		LL	UL	Unit
No conflict between ADG and ICAO	Wing span	52	65	m
	Tail Height	18,3	20,1	m
	OMGW span	9	14	m
	s_{TOFL}	1800	3000	m
	V_{APP}	141	165	kt

Figure 5.1 Screenshot: Reverse Engineering.xlsm – Data – SKYbrary

5.1.2 Airfoil

The reverse engineering results give a value for the maximum lift coefficient for take-off and landing. If the user wants to verify this value through a theoretical way, certain data has to be integrated. This data contains fixed values and formulas. It is important for the user to know that the theoretical approach is difficult to execute. The reason is that information about an airplanes' aerodynamics is hard to get because these date are classified information. Thus the theoretical approach will not be a good reference in every case. It is important to possess accurate information.

The formulas in this section are based on **Bhatia 2010**. In this work, every diagram is plotted and approached by equations. These equations are used in the Excel file in order to get the correct data. This section will explain briefly which formulas are used and how Excel is able to find and use this information.

The airfoil data is used in the '4) Verification' tab. The working of this tab is described in section 5.6. When reading this section, it will be clear that a lot of data is needed in order to perform the verification calculations. The first unknown parameters occur in equation (5.63). This formula shows the following unknown parameters:

$\left(c_{L,\max}\right)_{base}$	The maximum lift coefficient of the base of an airoil.
$\Delta_1 c_{L,max}$	A correction term for taking into account the airfoil camber and the position of maximum camber.
$\Delta_2 c_{L,max}$	A correction term for taking into account the position of the maximum thickness (if this is not 30%).
$\Delta_3 c_{L,max}$	A correction term for taking into account the Reynolds number (if this is not equal to $9 \cdot 10^6$).

Starting with the maximum lift coefficient of the base of an airoil, its relation with the leading edge sharpness parameter *Δy* and the position of the maximum thickness is shown in Figure 5.2. The leading edge sharpness parameter is the distance of the thickness of the airfoil between the 0,15%c and the 6,0%c point, expressed in % chord.

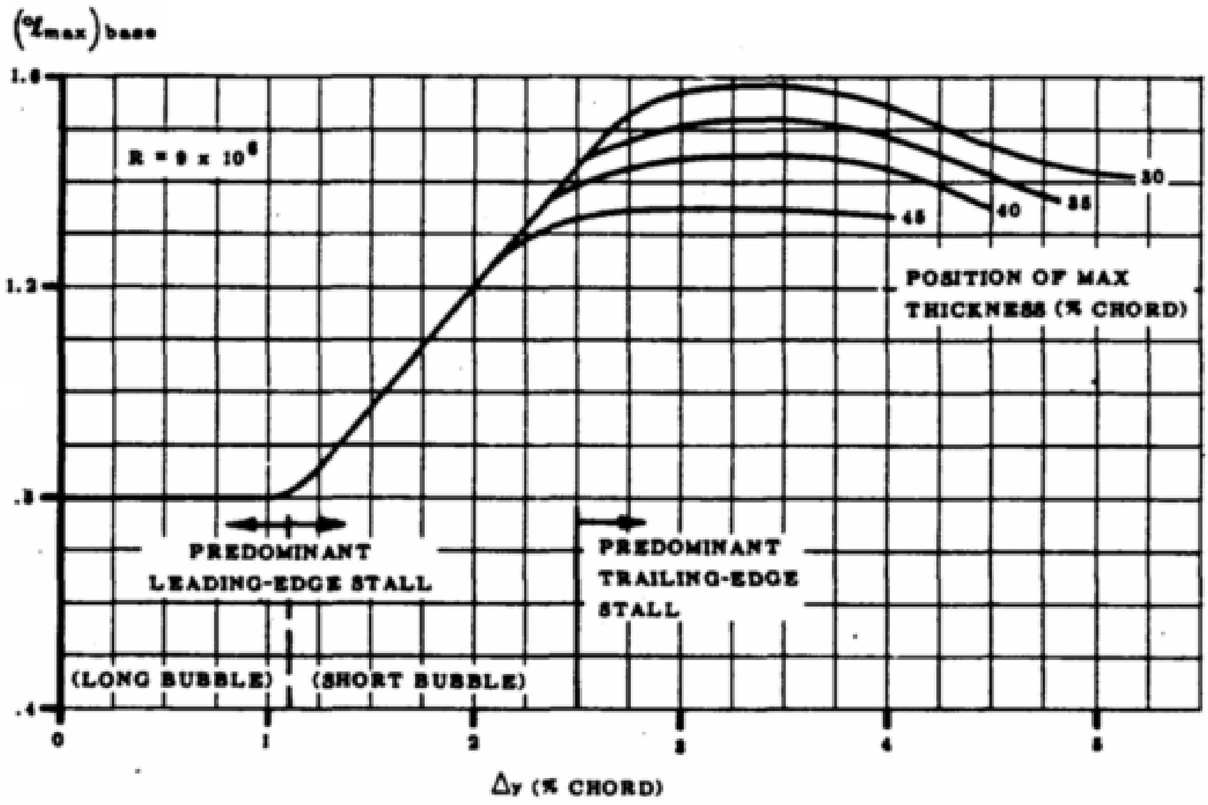


Figure 5.2 $(c_{L,max})_{base}$ in function of the leading edge sharpness parameter and the position of the maximum thickness (for Re = $9 \cdot 10^6$) (**DATCOM 1978**)

The leading edge sharpness parameter *Δy* can easily be found for NACA airfoils. The user selects the airfoil type and the *Δy* is automatically calculated according to Table 5.4. If the airfoil of the aeroplane is not in this table, the user has the possibility to add own values.

Table 5.4 *Δy*-parameter for known NACA airfoils (determined from **DATCOM 1978**)

Airfoil type	*Δy/(t/c)*
NACA 4 digit	26,0
NACA 5 digit	26,0
NACA 63 series	22,0
NACA 64 series	21,3
NACA 65 series	19,3
NACA 66 series	18,3

The following formulas (according to **Bhatia 2010**) are used in the Excel file. The maximum lift coefficient of the base of an airoil is determined using these equations which are fully integrated in the Excel file. This is done with logical programming using 'if' and 'and' commands to match the correct position of maximum thickness and leading edge sharpness parameter.

Position of maximum thickness x_t/c = 30%c:

$$\left(c_{L,\max}\right)_{base} = \begin{cases} 1 & ,\Delta y \leq 1 \\ \begin{gathered} -0{,}00280645\Delta y^6 + 0{,}04815713\Delta y^5 \\ -0{,}29947023\Delta y^4 + 0{,}78109758\Delta y^3 \\ -0{,}69599428\Delta y^2 + 0{,}18388579\Delta y \\ +0{,}79719724 \end{gathered} & ,\Delta y > 1 \end{cases} \tag{5.1}$$

Position of maximum thickness x_t/c = 35%c:

$$\left(c_{L,\max}\right)_{base} = \begin{cases} 1 & ,\Delta y \leq 1 \\ \begin{gathered} -0{,}00482159\Delta y^6 + 0{,}07449298\Delta y^5 \\ -0{,}42404095\Delta y^4 + 1{,}0401057\Delta y^3 \\ -0{,}93185333\Delta y^2 + 0{,}26204257\Delta y \\ +0{,}79427037 \end{gathered} & ,\Delta y > 1 \end{cases} \tag{5.2}$$

Position of maximum thickness x_t/c = 40%c:

$$\left(c_{L,\max}\right)_{base} = \begin{cases} 1 & ,\Delta y \leq 1 \\ \begin{gathered} -0{,}0070937\Delta y^6 + 0{,}1022105\Delta y^5 \\ -0{,}054740535\Delta y^4 + 1{,}27849448\Delta y^3 \\ -1{,}13177162\Delta y^2 + 0{,}32243359\Delta y \\ +0{,}79237084 \end{gathered} & ,\Delta y > 1 \end{cases} \tag{5.3}$$

Position of maximum thickness x_t/c = 45%c:

$$\left(c_{L,\max}\right)_{base} = \begin{cases} 1 & ,\Delta y \leq 1 \\ \begin{gathered} -0{,}01000105\Delta y^6 + 0{,}13630095\Delta y^5 \\ -0{,}68846582\Delta y^4 + 1{,}52817615\Delta y^3 \\ -1{,}31937651\Delta y^2 + 0{,}37173126\Delta y \\ +0{,}79122424 \end{gathered} & ,\Delta y > 1 \end{cases} \tag{5.4}$$

In most cases, the position of maximum thickness will differ from 30%, 35%, 40% or 45%. For example, the actual position of maximum thickness is 42%. Excel will compare this value with 30%, 35%, 40% and 45%. The two closest values are slected, here this is 40% and 45%. Each value is assigned to a formula, which is respectively (5.3) and (5.4). Next, linear interpolation is executed in order to achieve the best possible value for the maximum lift coefficient. All of this is realized using logic commands such as: if, else, and, greater then.

The next parameter is the correction term that takes the airfoils' camber and the position of maximum camber into account. **DATCOM 1978** provides four different graphs for this correction term. Each graph represents the correction term for a different maximum camber position.

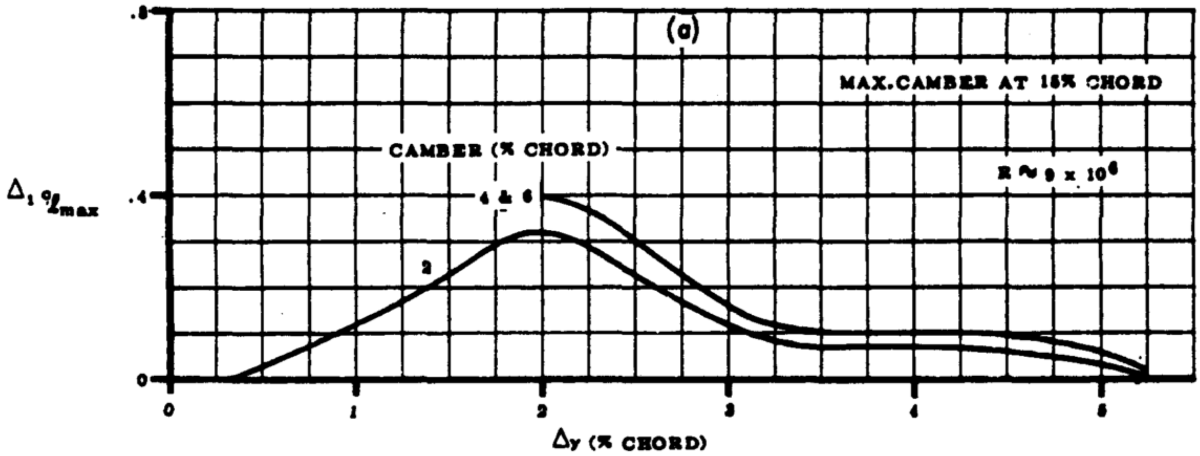


Figure 5.3 Correction term $\Delta_1 c_{L,max}$ considering the camber and the maximum camber at 15%c (**DATCOM 1978**)

Position of maximum camber 15%c and camber is 2%c:

$$\Delta_1 c_{L,\max} = \begin{cases} \begin{array}{c} -0{,}07774815\Delta y^6 + 0{,}37623248\Delta y^5 \\ -0{,}60831909\Delta y^4 + 0{,}29122922\Delta y^3 \\ +0{,}19328868\Delta y^2 - 0{,}05669713\Delta y \\ -0{,}00005284 \end{array} & ,\Delta y \leq 2 \\ \begin{array}{c} -0{,}00256675\Delta y^6 + 0{,}06762787\Delta y^5 \\ -0{,}72694424\Delta y^4 + 4{,}0386559\Delta y^3 \\ -12{,}09771291\Delta y^2 + 18{,}23870146\Delta y \\ -10{,}44224443 \end{array} & ,\Delta y > 2 \end{cases} \tag{5.5}$$

Position of maximum camber 15%c and camber is 4%c to 6%c:

$$\begin{aligned} \Delta_1 c_{L,\max} = & -0{,}00563605\Delta y^6 + 0{,}14130094\Delta y^5 - 1{,}45322643\Delta y^4 \\ & +7{,}78800147\Delta y^3 - 22{,}73899956\Delta y^2 + 33{,}8887824\Delta y \\ & -19{,}64527317 \end{aligned} \tag{5.6}$$

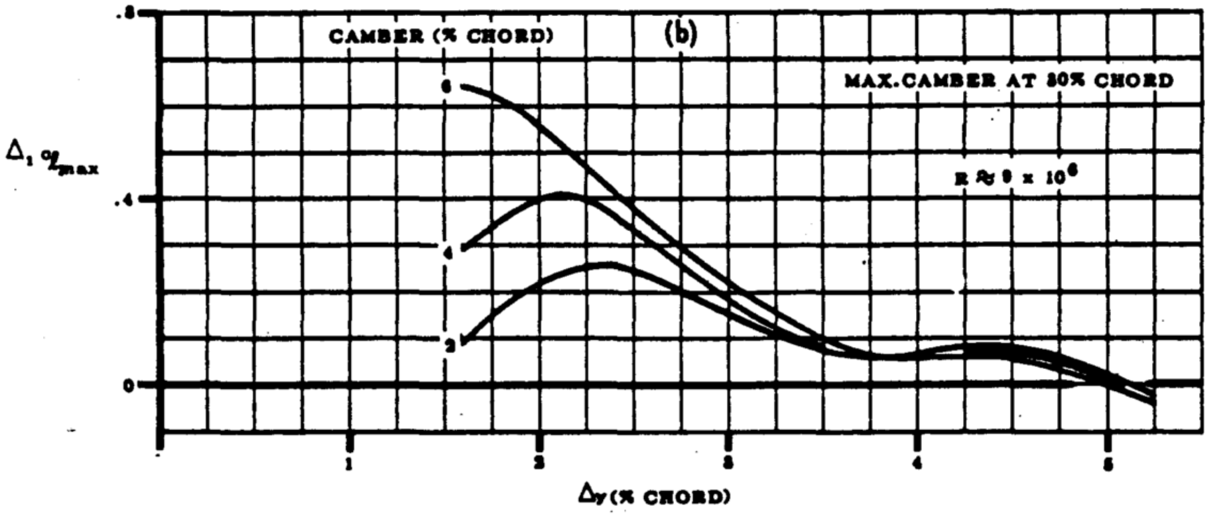


Figure 5.4 Correction term $\Delta_1 c_{L,max}$ considering the camber and the maximum camber at 30%c (**DATCOM 1978**)

Position of maximum camber 30%c and camber is 2%c:

$$\Delta_1 c_{L,\max} = 0{,}01073051\Delta y^6 - 0{,}22044538\Delta y^5 + 1{,}80311397\Delta y^4 \\ -7{,}44203294\Delta y^3 + 16{,}12094052\Delta y^2 - 17{,}09229486\Delta y \\ +6{,}97096640 \tag{5.7}$$

Position of maximum camber 30%c and camber is 4%c:

$$\Delta_1 c_{L,\max} = \begin{cases} 0{,}14032593\Delta y^6 - 2{,}27227351\Delta y^5 \\ +15{,}11676356\Delta y^4 - 52{,}6216939\Delta y^3 \\ +100{,}43734891\Delta y^2 - 99{,}04736377\Delta y \\ +39{,}57833727 & ,\Delta y \le 3{,}25 \\ -0{,}11567407\Delta y^6 + 3{,}02353505\Delta y^5 \\ -32{,}706416\Delta y^4 + 187{,}31615568\Delta y^3 \\ -5998{,}78383624\Delta y^2 + 1012{,}54322009\Delta y \\ -707{,}25981296 & ,\Delta y > 3{,}25 \end{cases} \tag{5.8}$$

Position of maximum camber 30%c and camber is 6%c:

$$\Delta_1 c_{L,\max} = \begin{cases} 0{,}01706667\Delta y^6 - 0{,}27076923\Delta y^5 \\ +1{,}71938462\Delta y^4 - 5{,}50270397\Delta y^3 \\ +9{,}14276552\Delta y^2 - 7{,}4776226\Delta y \\ +3{,}02301866 & ,\Delta y \le 3{,}25 \\ -0{,}04551111\Delta y^6 + 1{,}27540513\Delta y^5 \\ -14{,}64013678\Delta y^4 + 88{,}13476939\Delta y^3 \\ -293{,}54103794\Delta y^2 + 512{,}88085675\Delta y \\ -367{,}15343645 & ,\Delta y > 3{,}25 \end{cases} \tag{5.9}$$

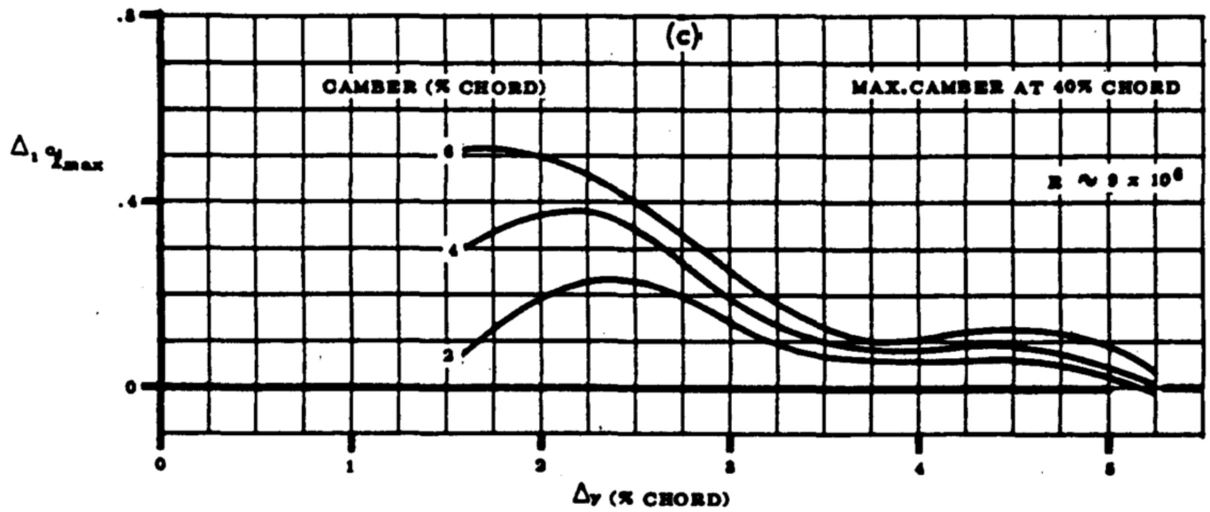


Figure 5.5 Correction term $\Delta_1 c_{L,max}$ considering the camber and the maximum camber at 40%c (**DATCOM 1978**)

Position of maximum camber 40%c and camber is 2%c:

$$\Delta_1 c_{L,\max} = 0{,}01051916\Delta y^6 - 0{,}218152\Delta y^5 + 1{,}80361567\Delta y^4 - 7{,}53960887\Delta y^3 + 16{,}60165778\Delta y^2 - 18{,}0064299\Delta y + 7{,}56777118 \quad (5.10)$$

Position of maximum camber 40%c and camber is 4%c:

$$\Delta_1 c_{L,\max} = 0{,}012659\Delta y^6 - 0{,}25826975\Delta y^5 + 2{,}09652033\Delta y^4 - 8{,}58472453\Delta y^3 + 18{,}48693421\Delta y^2 - 19{,}69531378\Delta y + 8{,}40319413 \quad (5.11)$$

Position of maximum camber 40%c and camber is 6%c:

$$\Delta_1 c_{L,\max} = 0{,}00711992\Delta y^6 - 0{,}15050618\Delta y^5 + 1{,}25405731\Delta y^4 - 5{,}22006422\Delta y^3 + 11{,}33891439\Delta y^2 - 12{,}25869283\Delta y + 5{,}71810516 \quad (5.12)$$

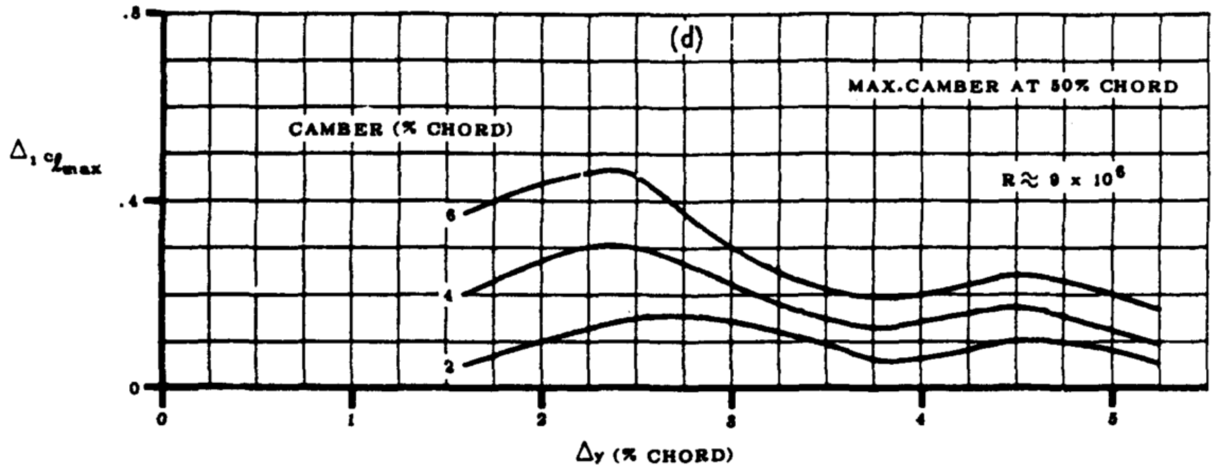


Figure 5.6 Correction term $\Delta_1 c_{L,max}$ considering the camber and the maximum camber at 50%c (**DATCOM 1978**)

Position of maximum camber 50%c and camber is 2%c:

$$\Delta_1 c_{L,\max} = \begin{cases} 0{,}03034074\Delta y^6 - 0{,}4157265\Delta y^5 \\ +2{,}3314416\Delta y^4 - 6{,}8870241\Delta y^3 \\ +11{,}31566756\Delta y^2 - 9{,}6807436\Delta y \\ +3{,}35293318 & ,\Delta y \leq 3{,}5 \\ -0{,}064474407\Delta y^6 + 1{,}76224274\Delta y^5 \\ -19{,}88139035\Delta y^4 + 118{,}43702273\Delta y^3 \\ -392{,}72462108\Delta y^2 + 686{,}96133483\Delta y \\ -494{,}97168447 & ,\Delta y > 3{,}5 \end{cases} \tag{5.13}$$

Position of maximum camber 50%c and camber is 4%c:

$$\Delta_1 c_{L,\max} = \begin{cases} -0{,}00686013\Delta y^6 + 0{,}08119216\Delta y^5 \\ -0{,}28547009\Delta y^4 + 0{,}00830193\Delta y^3 \\ -1{,}88581602\Delta y^2 + 3{,}33426764\Delta y \\ -1{,}87361888 & ,\Delta y \leq 4 \\ -1{,}08088889\Delta y^6 + 29{,}18399992\Delta y^5 \\ -327{,}288888\Delta y^4 + 1951{,}33332802\Delta y^3 \\ -6523{,}1802044\Delta y^2 + 11592{,}85763487\Delta y \\ -8556{,}87997644 & ,\Delta y > 4 \end{cases} \tag{5.14}$$

Position of maximum camber 50%c and camber is 6%c:

$$\Delta_1 c_{L,\max} = \begin{cases} 0{,}0853333\Delta y^5 - 1{,}17333333\Delta y^4 \\ +5{,}84\Delta y^3 - 13{,}72666666\Delta y^2 \\ +15{,}66466666\Delta y - 6{,}67 & ,\Delta y \leq 2{,}5 \\ 0{,}02691068\Delta y^6 - 0{,}60612569\Delta y^5 \\ +5{,}583559644\Delta y^4 - 26{,}9538523\Delta y^3 \\ +72{,}10649662\Delta y^2 - 101{,}90668299\Delta y \\ +60{,}21820782 & ,\Delta y > 2{,}5 \end{cases} \tag{5.15}$$

For the first correction term, Excel will apply linear interpolation for the position of maximum camber. This is already integrated in the Excel file and thus not important for the user in order to perform the verification.

The pogramming to assure that Excel will calculate and select the right value for the second correction term $\Delta_1 c_{L,max}$ is more complex. The interpolation for $\Delta_1 c_{L,max}$ works as follows. There is an inner- and outer interpolation. This means that two steps are required to perform the interpolations. The first step is to find the limits of the position of the maximum camber (equations (5.16) and (5.17)). This is done by rounding the real position of maximum camber to its nearest value that is provided with formulas (thus 15%c - 30%c - 40%c - 50%c). The same is done for the camber (thus 2%c - 4%c - 6%c) but twice since there are two values for the maximum camber position (equations (5.18) and (5.19)). In the next step, Excel will search for the associated value that matches for the upper and lower limit for the maximum camber position and camber. There are now four values for the first correction term (Table 5.5). The inner interpolation is done for the camber and has to be performed twice since there are two pairs of limits for the camber. The outer interpolation is then executed, for the position of maximum camber, to determine the final value for the maximum camber position.

$$= IF(FLOOR(x_{y\_c,max}; 5) < 15; 15; IF(FLOOR(x_{y\_c,max}; 5) = 15; FLOOR(x_{y\_c,max}; 5); IF(FLOOR(x_{y\_c,max}; 10) = 20; 15; FLOOR(x_{y\_c,max}; 10)))) \tag{5.16}$$

$$= IF(CEILING(x_{y\_c,max}; 5) < 15; 15; IF(CEILING(x_{y\_c,max}; 5) = 15; CEILING(x_{y\_c,max}; 5); IF(CEILING(x_{y\_c,max}; 10) = 20; 30; CEILING(x_{y\_c,max}; 10)))) \tag{5.17}$$

$$= FLOOR(y_{c,max}/c; 2) \tag{5.18}$$

$$= CEILING(y_{c,max}/c; 2) \tag{5.19}$$

Table 5.5 shows the result from the tab Data when the maximum camber is set at 28%c and the camber is set at 2,5%c. The separate results are listed in the right bottom of the table. These results are then interpolated twice.

Table 5.5 Reverse Engineering.xlsm – Data – Airfoil – $\Delta_1 c_{L,max}$

Maximum camber at (%c)	Camber (%c)	$\Delta_1 c_{L,max}$
	2	0,12
15	4	0,23
	6	0,23
	2	0,20
30	4	0,26
	6	0,29
	2	0,19
40	4	0,27
	6	0,33
	2	0,15
50	4	0,27
	6	0,37
Using equation (5.16)	2 (Using equation (5.18))	0,12
15	4 (Using equation (5.19))	0,23
Using equation (5.17)	2 (Using equation (5.18))	0,20
30	4 (Using equation (5.19))	0,26

The second correction term takes the position of the maximum thickness into account, at least if this differs from 30%. When the the position of the maximum thickness is 30%, the correction term $\Delta_2 c_{L,max}$ is zero. This factor is determined by the leading edge sharpness parameter and the position of the maximum thickness. Their mutual relation is shown in Figure 5.7.

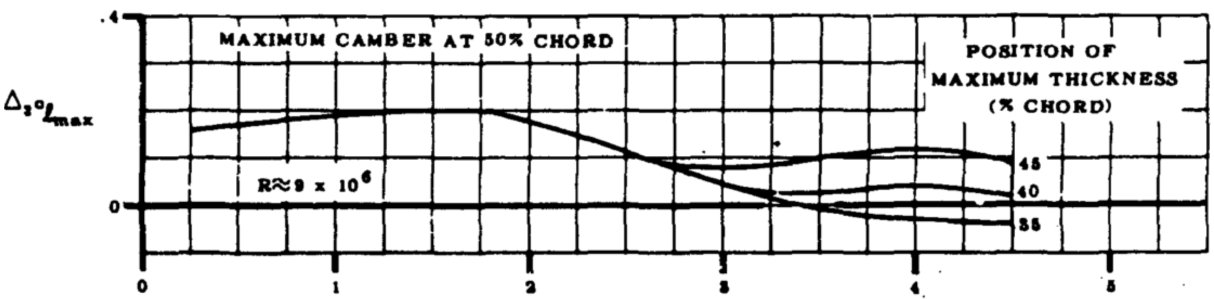


Figure 5.7 Correction term $\Delta_2 c_{L,max}$ considering the position of maximum thickness (**DATCOM 1978**)

Position of maximum thickness 35%c:

$$\begin{aligned}\Delta_2 c_{L,\max} = &-0{,}00081727\Delta y^6 + 0{,}00737469\Delta y^5 - 0{,}0095814\Delta y^4 \\ &+0{,}06687349\Delta y^3 - 0{,}163606\Delta y^2 + 0{,}07214101\Delta y \\ &-0{,}17056674\end{aligned} \tag{5.20}$$

Position of maximum thickness 40%c:

$$\Delta_2 c_{L,\max} = \begin{cases} -0{,}0044199\Delta y^6 + 0{,}0512218\Delta y^5 \\ -0{,}21198254\Delta y^4 + 0{,}3791454\Delta y^3 \\ -0{,}31722704\Delta y^2 + 0{,}1574402\Delta y \\ -0{,}135531058 & , \Delta y < 3{,}75 \\ -0{,}08\Delta y^2 + 0{,}64\Delta y - 1{,}245 & , \Delta y \geq 3{,}75 \end{cases} \tag{5.21}$$

Position of maximum thickness 45%c:

$$\Delta_2 c_{L,\max} = \begin{cases} 0{,}01167895\Delta y^6 - 0{,}08328495\Delta y^5 \\ +0{,}22023932\Delta y^4 - 0{,}29183069\Delta y^3 \\ +0{,}20078918\Delta y^2 - 0{,}02553077\Delta y \\ +0{,}15761818 & , \Delta y < 2{,}75 \\ 0{,}01167895\Delta y^6 - 0{,}08328495\Delta y^5 \\ +5{,}45207978\Delta y^4 - 24{,}15667757\Delta y^3 \\ +60{,}06275552\Delta y^2 - 79{,}76227484\Delta y \\ +44{,}43654240 & , \Delta y \geq 2{,}75 \end{cases} \tag{5.22}$$

Excel will search which values of the three positions of maximum thickness are the nearest to the real value. Then, a linear interpolation is executed to determine the second correction term. The integration in Excel is analog to the previous two correction terms.

The last correction term $\Delta_3 c_{L,max}$ will take the Reynolds number into account. The reference Reynolds number is $9 \cdot 10^6$. The Reynolds number that occurs at the mean aerodynamic chord is calculated with the following equation:

$$Re = \frac{\rho \cdot V_{APP} \cdot c_{MAC}}{\mu} \tag{5.23}$$

$$\mu = 1{,}458 \cdot 10^6 \cdot \frac{T_L^{1,5}}{T_L + 110{,}4} \tag{5.24}$$

Re	Reynolds number
ρ	Local air density [kg/m³]
V_{APP}	Approach speed [m/s]
c_{MAC}	Mean aerodynamic chord [m]
μ	Dynamic viscosity [kg/(m s)]
T_L	Landing temperature [K]

The mutual relation between the correction term and the Reynolds number in function of the leading edge sharpness parameter is shown in Figure 5.8:

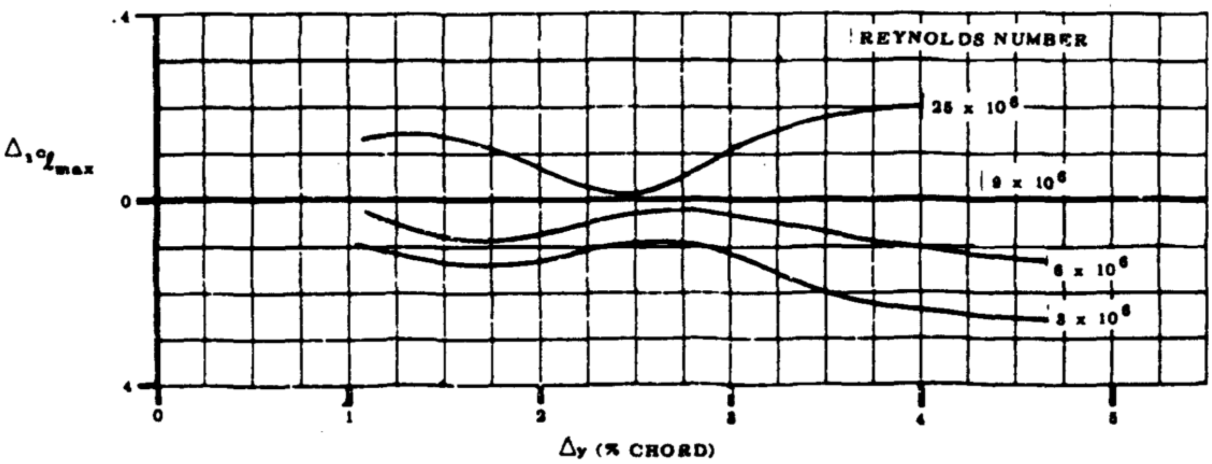


Figure 5.8 Correction term $\Delta_3 c_{L,max}$ considering the Reynolds number (**DATCOM 1978**)

To integrate these curves into the Excel file, the following equations are applied:

Reynolds number $Re = 3 \cdot 10^6$:

$$\Delta_3 c_{L,\max} = \begin{cases} \begin{array}{c} 0{,}14222222\Delta y^6 - 1{,}73784616\Delta y^5 \\ +8{,}64683762\Delta y^4 - 22{,}49123547\Delta y^3 \\ +32{,}43006611\Delta y^2 - 24{,}73784737\Delta y \\ +7{,}71405305 \end{array} & ,\Delta y < 3 \\ \begin{array}{c} -0{,}14222222\Delta y^6 + 3{,}37558975\Delta y^5 \\ -33{,}23760692\Delta y^4 + 173{,}73496547\Delta y^3 \\ -508{,}23048689\Delta y^2 + 788{,}40282126\Delta y \\ -506{,}43528965 \end{array} & ,\Delta y \geq 3 \end{cases} \tag{5.25}$$

Reynolds number $Re = 6 \cdot 10^6$:

$$\Delta_3 c_{L,\max} = \begin{cases} \begin{array}{c} 0{,}04551111\Delta y^6 - 0{,}50346667\Delta y^5 \\ +2{,}20526496\Delta y^4 - 4{,}93189744\Delta y^3 \\ +6{,}13388128\Delta y^2 - 4{,}2930345\Delta y \\ +1{,}34372727 \end{array} & ,\Delta y < 3 \\ \begin{array}{c} 0{,}01896296\Delta y^6 - 0{,}42338462\Delta y^5 \\ +3{,}90245015\Delta y^4 - 18{,}98365193\Delta y^3 \\ +51{,}34259628\Delta y^2 - 73{,}18781206\Delta y \\ +42{,}96881902 \end{array} & ,\Delta y \geq 3 \end{cases} \tag{5.26}$$

Reynolds number $Re = 9 \cdot 10^6$:

$$\Delta_3 c_{L,\max} = 0 \tag{5.27}$$

Reynolds number $Re = 25 \cdot 10^6$:

$$\Delta_3 c_{L,\max} = \begin{cases} \begin{array}{c} 0,17066667\Delta y^6 - 1,79199999\Delta y^5 \\ +7,73333331\Delta y^4 - 17,41333328\Delta y^3 \\ +21,27599993\Delta y^2 - 13,18466662\Delta y \\ +3,33 \end{array} & ,\Delta y < 2,5 \\ \begin{array}{c} -0,07111111\Delta y^6 + 1,24964103\Delta y^5 \\ -8,88341881\Delta y^4 + 32,36944057\Delta y^3 \\ -62,73926381\Delta y^2 + 59,65733744\Delta y \\ -20,45147001 \end{array} & ,\Delta y \geq 2,5 \end{cases} \tag{5.28}$$

Also for this correction term, linear interpolation is applied the same way as the other correction terms are interpolated.

Ones these parameters are all known, the maximum lift coefficient for a clean airfoil is calculated in the '4) Verification' tab with equation (5.63). The next formula that requires airfoil data is equation (5.64) and is used to calculate the maximum lift coefficient for a clean wing. This formula shows the following unknown parameters:

$\frac{C_{L,max}}{c_{L,max}}$	A correction term that takes the sweep angle into account.
$\Delta C_{L,max}$	A correction term that takes the Mach number for approach into account.

The lift coefficient decreases with increasing wing sweep angle for airfoils with a rounded leading edge. Airfoils with a sharp leading edge will show an increase of the lift coefficient due to the creation of vortices. The relation between this correction factor, the leading edge sharpness parameter and the leading edge sweep angle is shown in Figure 5.9.

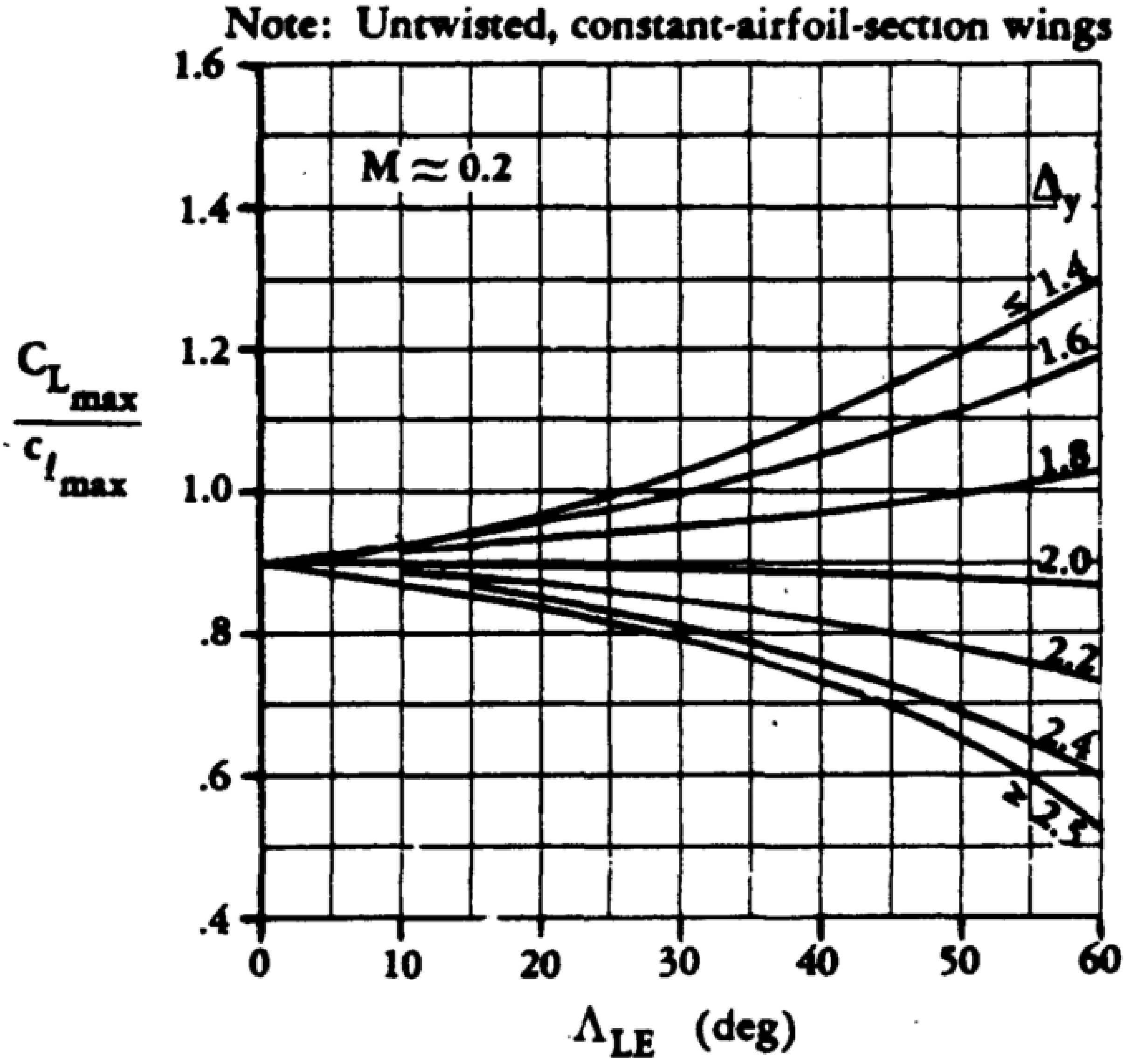


Figure 5.9 Maximum lift of tapered wings with a high aspect ratio in subsonic speeds (**DATCOM 1978**)

To integrate this graph in the Excel-file, an equation is build-up for each leading edge sharpness parameter. The equation describes the correction term as a function of the leading edge sweep angle in degrees.

Leading edge sharpness parameter $\Delta y \leq 1{,}4$:

$$\frac{C_{L,max}}{c_{L,max}} = 0{,}0000839\varphi_{LE}^2 + 0{,}00169643\varphi_{LE} + 0{,}89785714 \tag{5.29}$$

Leading edge sharpness parameter $\Delta y = 1{,}6$:

$$\frac{C_{L,max}}{c_{L,max}} = 0{,}00005\varphi_{LE}^2 + 0{,}001785714286\varphi_{LE} + 0{,}9 \tag{5.30}$$

Leading edge sharpness parameter Δy = 1,8:

$$\frac{C_{L,max}}{c_{L,max}} = 0{,}000000000625\varphi_{LE}^5 + 0{,}000000082386\varphi_{LE}^4 \\ -0{,}000003532197\varphi_{LE}^3 + 0{,}00006751894\varphi_{LE}^2 \\ +0{,}001052651506\varphi_{LE} + 0{,}90007034637 \tag{5.31}$$

Leading edge sharpness parameter Δy = 2,0:

$$\frac{C_{L,max}}{c_{L,max}} = -0{,}0000000114\varphi_{LE}^4 + 0{,}0000013636\varphi_{LE}^3 \\ -0{,}0000564394\varphi_{LE}^2 + 0{,}004318182\varphi_{LE} \\ +0{,}9000108225 \tag{5.32}$$

Leading edge sharpness parameter Δy = 2,2:

$$\frac{C_{L,max}}{c_{L,max}} = -0{,}000000017\varphi_{LE}^4 + 0{,}0000021843\varphi_{LE}^3 \\ -0{,}0011263258\varphi_{LE}^2 + 0{,}000568622\varphi_{LE} \\ +0{,}89995671 \tag{5.33}$$

Leading edge sharpness parameter Δy = 2,4:

$$\frac{C_{L,max}}{c_{L,max}} = -0{,}0000000379\varphi_{LE}^4 + 0{,}0000042677\varphi_{LE}^3 \\ -0{,}0002159091\varphi_{LE}^2 + 0{,}0006814574\varphi_{LE} \\ +0{,}9001948052 \tag{5.34}$$

Leading edge sharpness parameter $\Delta y \geq$ 2,5:

$$\frac{C_{L,max}}{c_{L,max}} = -0{,}000000017\varphi_{LE}^4 + 0{,}0000005177\varphi_{LE}^3 \\ -0{,}0000304924\varphi_{LE}^2 + 0{,}0026756854\varphi_{LE} \\ +0{,}8998376623 \tag{5.35}$$

This correction term is directly calculated with a formula. To integrate this in the Excel file, the program has to know which value it should use. This is done using logical commands and one linear interpolation.

The final correction term takes the approach Mach number into account. The larger the Mach number for approach, the smaller the final lift coefficient will be. If the Mach number is equally or smaller than 0,2 the correction term $\Delta C_{L,max}$ is zero. This term is also influenced by the leading edge sweep angle and leading edge sharpness parameter. Figure 5.10 shows four graphs for leading edge sweep angles of respectively 0°, 20°, 40° and 60°. The decrease of the lift coefficient gets larger if the leading edge sharpness parameter rises (except for a sweep angle of 60° in combination with a leading edge sharpness parameter of 2,0).

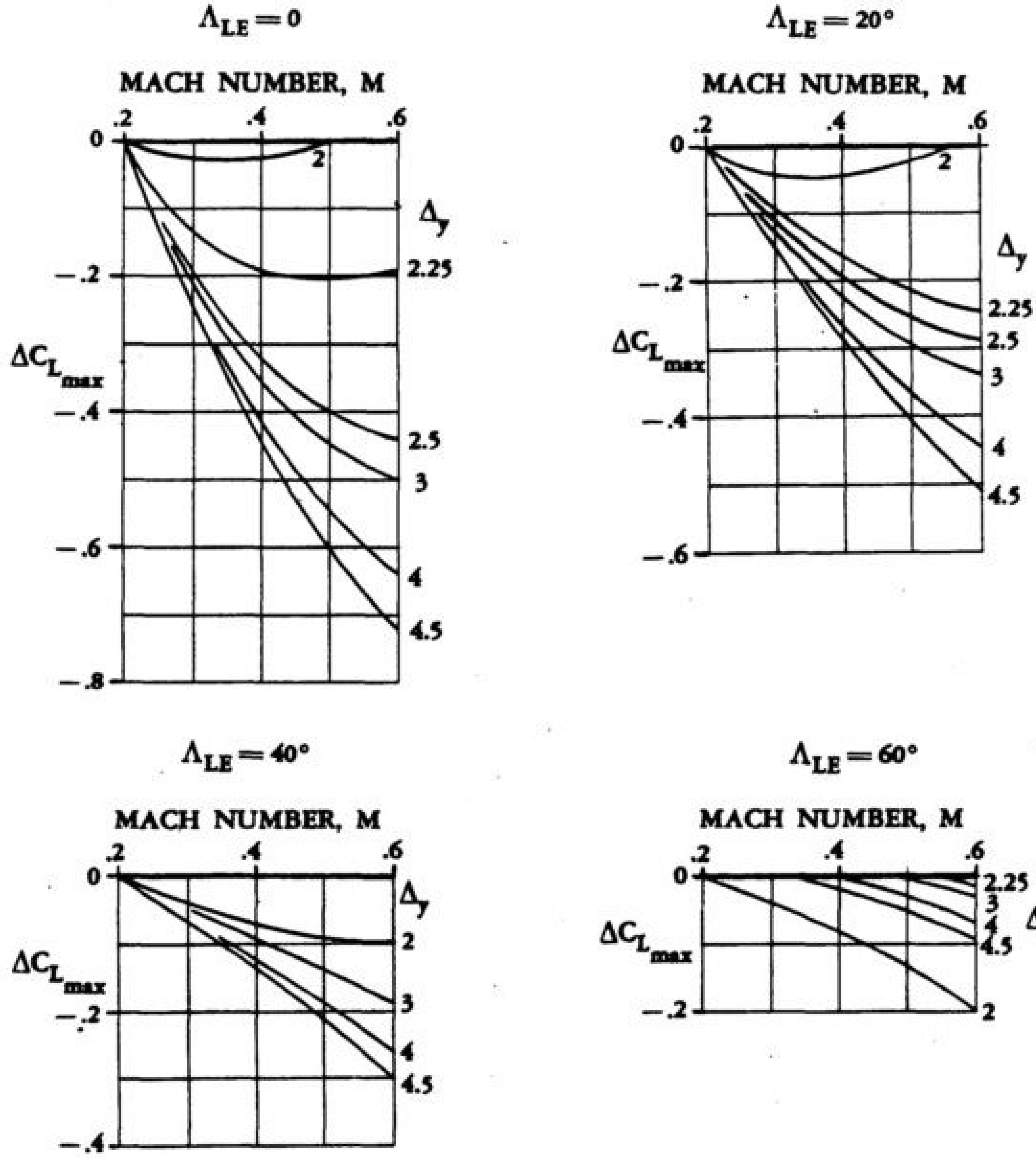


Figure 5.10 Correction term $\Delta C_{L,max}$ considering the approach Mach number for tapered wings with high aspect ratio (**DATCOM 1978**)

To integrate this in the Excel file, there must be an equation for every leading edge sweep angle and leading edge sharpness parameter. Each equation gives a solution for the correction term $\Delta C_{L,max}$ in function of the approach Mach number.

Leading edge sweep angle $\varphi_{LE} = 0°$:

$\Delta y = 2$
$$\Delta C_{L,max} = 0{,}702 M_{APP}^3 + 0{,}568 M_{APP}^2 - 0{,}658 M_{APP} + 0{,}103 \tag{5.36}$$

$\Delta y = 2{,}25$
$$\Delta C_{L,max} = 22{,}5 M_{APP}^4 - 39{,}5 M_{APP}^3 + 27{,}3 M_{APP}^2 - 8{,}96 M_{APP} + 0{,}98 \tag{5.37}$$

$\Delta y = 2{,}5$ $$\Delta C_{L,max} = -1{,}67M_{APP}^3 + 4{,}5M_{APP}^2 - 3{,}83M_{APP} + 0{,}6 \tag{5.38}$$

$\Delta y = 3$ $$\Delta C_{L,max} = -8{,}33M_{APP}^4 + 11{,}7M_{APP}^3 - 2{,}92M_{APP}^2 + 2{,}32M_{APP} + 0{,}5 \tag{5.39}$$

$\Delta y = 4$ $$\Delta C_{L,max} = -33{,}3M_{APP}^4 + 55M_{APP}^3 - 30{,}2M_{APP}^2 + 4{,}6M_{APP} - 0{,}1 \tag{5.40}$$

$\Delta y = 4{,}5$ $$\Delta C_{L,max} = -49{,}2M_{APP}^4 + 78{,}7M_{APP}^3 - 42{,}8M_{APP}^2 + 7{,}23M_{APP} - 0{,}286 \tag{5.41}$$

Leading edge sweep angle $\varphi_{LE} = 20°$:

$\Delta y = 2$ $$\Delta C_{L,max} = -5{,}83M_{APP}^4 + 5{,}67M_{APP}^3 + 0{,}0583M_{APP}^2 - 1{,}13M_{APP} + 0{,}187 \tag{5.42}$$

$\Delta y = 2{,}25$ $$\Delta C_{L,max} = -4{,}17M_{APP}^4 + 5{,}83M_{APP}^3 - 1{,}96M_{APP}^2 - 0{,}758M_{APP} + 0{,}19 \tag{5.43}$$

$\Delta y = 2{,}5$ $$\Delta C_{L,max} = 20{,}8M_{APP}^4 - 32{,}5M_{APP}^3 + 19{,}3M_{APP}^2 - 5{,}93M_{APP} + 0{,}64 \tag{5.44}$$

$\Delta y = 3$ $$\Delta C_{L,max} = -8{,}33M_{APP}^4 + 13{,}3M_{APP}^3 - 6{,}42M_{APP}^2 + 0{,}0167M_{APP} + 0{,}16 \tag{5.45}$$

$\Delta y = 4$ $$\Delta C_{L,max} = -4{,}17M_{APP}^4 + 5{,}83M_{APP}^3 - 1{,}96M_{APP}^2 - 1{,}26M_{APP} + 0{,}29 \tag{5.46}$$

$\Delta y = 4{,}5$ $$\Delta C_{L,max} = 20{,}8M_{APP}^4 - 32{,}5M_{APP}^3 + 18{,}8M_{APP}^2 - 6{,}08M_{APP} + 0{,}69 \tag{5.47}$$

Leading edge sweep angle $\varphi_{LE} = 40°$:

$\Delta y = 2$ $$\Delta C_{L,max} = -3{,}33M_{APP}^4 + 6{,}33M_{APP}^3 - 3{,}67M_{APP}^2 + 0{,}467M_{APP} + 0{,}008 \tag{5.48}$$

$\Delta y = 3$ $$\Delta C_{L,max} = -5{,}83M_{APP}^4 + 8{,}83M_{APP}^3 - 4{,}74M_{APP}^2 + 0{,}592M_{APP} + 0{,}01 \tag{5.49}$$

$\Delta y = 4$ $$\Delta C_{L,max} = -8{,}75M_{APP}^4 + 14{,}4M_{APP}^3 - 8{,}61M_{APP}^2 + 1{,}56M_{APP} - 0{,}068 \tag{5.50}$$

$$\Delta y = 4{,}5 \qquad \Delta C_{L,max} = -13{,}3M_{APP}^4 + 20{,}2M_{APP}^3 - 11{,}1M_{APP}^2 + 1{,}92M_{APP} - 0{,}08 \tag{5.51}$$

Leading edge sweep angle $\varphi_{LE} = 60°$:

$$\Delta y = 2 \qquad \Delta C_{L,max} = -2{,}84M_{APP}^2 + 2{,}98M_{APP} - 0{,}781 \tag{5.52}$$

$$\Delta y = 2{,}25 \qquad \Delta C_{L,max} - 0{,}97M_{APP}^2 + 0{,}825M_{APP} - 0{,}175 \tag{5.53}$$

$$\Delta y = 3 \qquad \Delta C_{L,max} - 0{,}45M_{APP}^2 + 0{,}115M_{APP} + 0{,}026 \tag{5.54}$$

$$\Delta y = 4 \qquad \Delta C_{L,max} = -5{,}29M_{APP}^3 + 7{,}25M_{APP}^2 - 3{,}58M_{APP} + 0{,}592 \tag{5.55}$$

$$\Delta y = 4{,}5 \qquad \Delta C_{L,max} = 1{,}25M_{APP}^4 - 2{,}42M_{APP}^3 + 1{,}24M_{APP}^2 - 0{,}621M_{APP} + 0{,}092 \tag{5.56}$$

Because this correction term is depending on three parameters, there are two linear interpolations required in order to determine the value that matches the real leading edge sharpness parameter and the leading edge sweep angle. The integration of the double linear interpolation in Excel, is analog to the integration for the correction term $\Delta_{1}c_{L,max}$. The following equations with logical functions are used:

$$= FLOOR(\varphi_{LE}; 20) \tag{5.57}$$

$$= CEILING(\varphi_{LE}; 20) \tag{5.58}$$

$$= IF(FLOOR(y; 0{,}25) = 2{,}25; 2{,}25; IF(FLOOR(y; 0{,}5) = 2{,}5; 2{,}5; IF(FLOOR(y; 0{,}5) = 4{,}5; 4{,}5; IF(FLOOR(y; 1) < 2; 2; FLOOR(y; 1))))) \tag{5.59}$$

$$= IF(CEILING(y; 0{,}25) = 2{,}25; 2{,}25; IF(CEILING(y; 0{,}5) = 2{,}5; 2{,}5; IF(CEILING(y; 0{,}5) = 4{,}5; 4{,}5; IF(CEILING(y; 1) > 4; 4; CEILING(y; 1))))) \tag{5.60}$$

Table 5.6 Reverse Engineering.xlsm – Data – Airfoil – $\Delta C_{L,max}$

φ_{LE}	Δy	$\Delta C_{L,max}$
0	2,00	-0,01
	2,25	-0,04
	2,50	-0,05
	3,00	-0,06
	4,00	-0,05
	4,50	-0,05
20	2,00	-0,01
	2,25	-0,02
	2,50	-0,03
	3,00	-0,03
	4,00	-0,04
	4,50	-0,04
40	2,00	-0,01
	2,25	-0,01
	2,50	-0,01
	3,00	-0,01
	4,00	-0,01
	4,50	-0,01
60	2,00	0,00
	2,25	0,00
	2,50	0,00
	3,00	0,00
	4,00	0,00
	4,50	-0,01
Using equation (5.57)	2,5 (Using equation (5.59))	-0,01
40	3 (Using equation (5.59)(5.60))	-0,01
Using equation (5.58)	2,5 (Using equation (5.59))	0,00
60	3 (Using equation (5.59)(5.60))	0,00

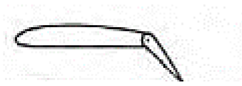
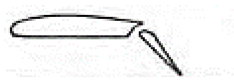
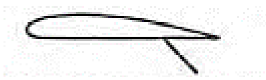
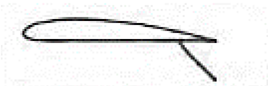
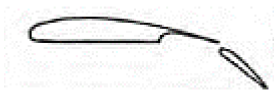
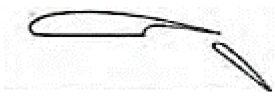
Now that every correction term is known, the maximum lift coefficient for a clean wing can be calculated using equation (5.64) in the tab '4) Verification'. Figure 5.20 shows the userinterface where all the correction terms and the maximum lift coefficient for a clean wing are represented.

5.1.3 High-lift Systems

The previous section explained which airfoil data are necessary to calculate the maximum lift coefficient of a clean wing in the tab '4) Verification'. The next step is to integrate high-lift systems. It should be possible for the user to select which trailing edge and leading edge high-lift systems are used. By doing this, Excel will calculate the effect on the total lift coefficient. Therefore, the contribution to the maximum lift coefficient from each type of high-lift systems should be known.

The contribution of trailing edge high-lift systems (flaps) is calculated with equation (5.65) (section 5.6.1). $\Delta c_{L,max,f}$ is the increase of the maximum lift coefficient of the clean wing produced by the flaps. It is a certain percentage of the maximum lift coefficient of the clean wing. The percentage is depending on the type of high-lift system as is shown in Table 5.7:

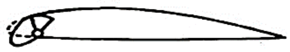
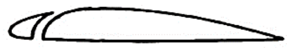
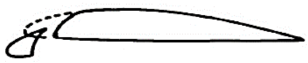
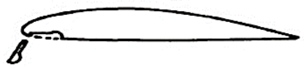
Table 5.7 Flap characteristics (**Stinton 1983**)

Description	Profile	Increase of lift coefficient
0,3c Plain flap deflected 45°		51%
0,3c Single slotted flap deflected 45°		53%
Double slotted flap[2]		98%
0,3c Split flap deflected 45°		67%
0,3c Split (Zap) flap hinged at 0,8c - deflected 45°		75%
0,3c Split (Zap) flap hinged at 0,9c - deflected 45°		80%
0,3c Fowler flap deflected 40°		119%
0,4c Fowler flap deflected 40°		140%

[2] The double slotted flap was not mentioned in **Stinton 1983**, therefore the value is taken from **Dubs 1987**.

The contribution of leading edge high-lift systems (slats) is calculated with equation (5.67). $\Delta c_{L,max,s}$ is determined the same way $\Delta c_{L,max,f}$ is; as a percentage of the maximum lift coefficient of the clean wing. These percentages are shown in Table 5.8.

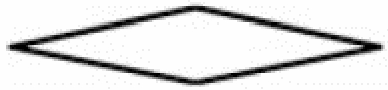
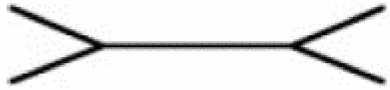
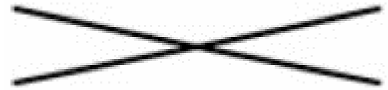
Table 5.8 Slat characteristics (**Stinton 1983**)

Description	Profile	Increase of lift coefficient
0,3c Nose flap deflected 30° - 40°		62%
Fixed slat forming a slot		37%
Handley Page automatic slat		43%
0,1c Kruger flap		46%

5.1.4 Winglets

In order to make the ‘4) Verification’ tab as detailed as possible, but still easy to handle, the user is able to choose different winglets with a certain size. This influences the effective aspect ratio. In section 5.6.2, it is explained how the maximum aerodynamic efficiency is calculated using this effective aspect ratio. If the winglet is a normal end plate, there is no additional data necessary to execute equation (5.70). When the winglet has another non-planar shape, external data is used to determine the effective aspect ratio, using equation (5.71). This data and winglet configurations are shown in Table 5.9.

Table 5.9 Span efficiency for various optimally loaded non-planar configurations (h/b = 0,2) (**Kroo 2005**)

Non-planar configuration		$k_{e,NP}$
V-wing		1,03
Diamond wing		1,05
X-flat wing		1,32
X-wing		1,33
Double wing		1,36
H-wing		1,38
End plate (k_{WL} = 2,13)		1,41
Quasi-closed C-wing		1,45
Box wing		1,46

5.2 Specifications and Reverse Engineering Results

The tab 'Specs + RE' contains the specifications and the reverse engineering results of the aeroplane. The idea of this tab is to serve as central tab for the user. Almost all the input, that has to be done by the user, happens here. It is the most important tab for the operator. Besides this, there is only few that has to be filled out by the user in other tabs. All fields in bold blue have to be filled out by the user. The bold red fields are results and should not be touched. Black fields are calculated or repeated values.

5.2.1 Aeroplane Specifications

There are three different parts with each its specific application it is used for. The first part of this tab is the 'Data to apply reverse engineering'. The user does research in order to find as much usefull aeroplane specifications as possible and fills out the bold blue fields. Note that there is the possibility to have some values unknown. This comes with the negative side-effect that if one value is unknown, another has to be known. If the take-off field length or the wing span is unknown or if both the landing field length and the approach speed is unknown; the user has to fill out the section SKYbrary in the tab 'Data'. This is simply done by choosing the correct aircraft category in all four dropdown menus.

To continue with the parameter status, if the cruise altitude or the cruise speed is unknown, a field will appear where the user is asked to fill out the lower limit and upper limit. Note that once these fields are filled out, it is impossible for the user to return to the state where the cruise speed or cruise altitude is known. The field is contains a formula which adapts to the parameter state. By filling out the limits, this formula gets overwritten and thus lost. The program is provided by a function that warns the user about this.

The type of range should also be chosen. The user can select the following range types: range for maximum payload, range for maximum amount of passengers, maximum range and a different range according to the payload-range diagram of the aeroplane. This input is linked to the payload. By changing the range type field, the program tells the user which payload mass has to be given. If the maximum range is selected, the user is asked to fill out the fuel capacity. In section 4.2.3 it is explained that there are two methods to calculate the specific fuel consumption if the maximum range is used. It is possible using the available fuel volume of the airplane or the payload mass and operating empty mass. The grey fields represent the upper and lower limits from SKYbrary data. An example of this part is shown in Figure 5.11.

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**1905**	m		
Approach speed	**Known**	V_{APP}	**75,10**	m/s	75,1	75,1
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2815**	m	2815	2815
Temperature above ISA (288,15K)		ΔT_{TO}	**15**	K		
Relative density		σ	**0,951**			
Maximum range		R	**7500**	NM		
Cruise Mach number		M_{CR}	**0,855**			
Wing area		S_W	**541**	m²		
Wing span	**Known**	b_W	**64,44**	m²	64,44	64,44
Aspect ratio		A	7,67			
Maximum take-off mass		m_{MTO}	**362870**	kg		
Payload mass (maximum range)		m_{PL}	**18000**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,050			
Maximum landing mass		m_{ML}	**260360**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,718			
Operating empty mass		m_{OE}	**183160**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,505			
Wing loading		m_{MTO}/S_W	670,5	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**253**	kN		
Total take-off thrust		T_{TO}	1012	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,284			
Bypass ratio		μ	**4,85**			
Available fuel volume		$V_{fuel,available}$	**204,35**	m³		

Figure 5.11 Screenshot: Reverse Engineering.xlsm – Specs + RE – Data to apply reverse engineering

The second part is the 'Data to optimize V/V_{md}'. This requires a value for the cruise altitude and cruise speed. When one of these values are unknown, two fields will appear to fill out the limits. The values for these parameters are used by the solver in Excel to minimize the squared sum of the differences in the reverse engineering. This is done by optimizing the V/V_{md} ratio. The ratio has a value between 1 and $\sqrt[4]{3}$ which are respectively the minimum drag speed and maximum range speed. A possible representation of this part is shown in Figure 5.12.

Data to optimize V/V_{md}

					LL	*UL*
Cruise speed	V_{CR}	**254**	m/s			
Cruise altitude	h_{CR}	**10622**	m			
Speed ratio	V/V_{md}	**1,000**	-		1	1,000

Figure 5.12 Screenshot: Reverse Engineering.xlsm – Specs + RE – Data to optimize V/V_{md}

The third part is the 'Data to execute the verification' which has only to be filled out if the user intends to do a verification on the reverse engineered values. This tab requires aerodynamic data from the wing. Only for the relative thickness there is the possibility to leave this value open. If it is unknown, the mean relative thickness is calculated using only the cruise Mach number (equation (5.61)). When the relative thickness is known and the user fills out the field, the formula in this field gets lost and there is no return to the unknown relative thickness state. The user gets informed about this issue by making use of a notification.

Data to execute the control

					Range
Sweep angle		φ_{25}	**37,5**	°	
Mean aerodynamic chord		c_{MAC}	**9,68**	m	
Position of maximum camber		$x_{(y\_c),max}$	**30**	%c	15 - 50 %c
Camber		$(y_c)_{max}/c$	**6**	%c	2 - 6 %c
Position of maximum thickness		$x_{t,max}$	**35**	%c	30 - 45 %c
Relative thickness	**Known**	t/c	**9,4**	%	
Taper		λ	**0,275**		

Figure 5.13 Screenshot: Reverse Engineering.xlsm – Specs + RE – Data to execute the verification

$$\frac{t}{c} = -0{,}0439 \cdot tan^{-1}(3{,}3450 \cdot M_{CR} - 3{,}0231) + 0{,}0986 \tag{5.61}$$

5.2.2 Reverse Engineering

The reverse engineering calculations are activated by clicking on the button 'Reverse Engineering'. This action starts the solver from Excel. This solver will adapt the following parameters if their status is set 'unknown': approach speed, take-off field length, span, cruise speed and altitude. The ratio V/V_{md} is adapted anyway. All these values are modified in such a way that the squared sum of the deviation for the following parameters are as small as possible:

Landing field length	s_{LFL}
Approach speed	V_{APP}
Take-off field length	s_{TOFL}
Wing span	b_W
Aspect ratio	A
Cruise speed	V_{CR}
Cruise altitude	h_{CR}

The results of the reverse engineering are listed next to the button, respectively: maximum lift coefficient for landing, maximum lift coefficient for take-off, maximum aerodynamic efficiency and the specific fuel consumption. This way, the user will immediately see the results after pushing the reverse engineering button. Furthermore, every specification that is "unknown" will be shown in bold red. An example of the reverse engineering in Excel is shown in the next figure.

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1905	1905	m	0,00%
Approach speed	V_{APP}	75,10	75,1	m/s	0,00%
Take-off field length	s_{TOFL}	2815	2815	m	0,00%
Span	b_W	64,44	64,44	m	0,00%
Aspect ratio	A	7,67	7,67		0,00%
Cruise speed	V_{CR}	254,3	252	m/s	-0,76%
Cruise altitude	h_{CR}	10622	11492	m	8,19%
Squared Sum					**6,77E-03**
Absolute maximum deviation					8,2%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**2,36**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,06**	
Maximum aerodynamic efficiency	E_{max}	**16,88**	
Specific fuel consumption (acc. to PL and OE)	SFC	**1,45E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**1,42E-05**	kg/N/s

Reverse Engineering

Figure 5.14 Screenshot: Reverse Engineering.xlsm – Specs + RE – Reverse Engineering

5.3 Maximum Lift Coefficient

This tab has two functions. Firstly, it calculates the maximum lift coefficient for landing and take-off, using equations (4.1) and (4.2). Secondly, the calculations for second segment and missed approach are executed as they are done in the preliminary sizing. By doing this, it is possible to create a matching chart which includes these values.

The only input the user has here is the choice of certification basis. Using FAR Part 25 will take the drag of the landing gear into account. JAR-25/CS-25 does the calculations with retracted landing gear and thus no additional drag.

Landing			
Landing field length	s_{LFL}	1905	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	75,10	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,72	
Wing loading at maximum take-off mass	m_{MTO}/S_W	670,5	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**2,36**	

Take-off			
Take-off field length	s_{TOFL}	2815	m
Temperatur above ISA (288,15K)	ΔT_{TO}	15	K
Relative density	σ	0,95	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,06**	

Figure 5.15 Screenshot: Reverse Engineering.xlsm – 1) C_Lmax

2nd Segment		
Aspect ratio	A	7,673
Lift coefficient, take-off	$C_{L,TO}$	1,43
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,017
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,037
Oswald efficiency factor; landing configuration	e	0,7
Glide ratio in take-off configuration	E_{TO}	9,05
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,187

Missed approach		
Lift coefficient, landing	$C_{L,L}$	1,40
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,015
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,050
Glide ratio in landing configuration	E_L	8,44
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,139

Figure 5.16 Screenshot: Reverse Engineering.xlsm – 1) C_Lmax

5.4 Maximum Aerodynamic Efficiency

In this tab called '2) E_max' the user does not have to fill out anything (unless the Oswald factor for a clean wing is not equal to 0,85). Everything that is necessary is already filled out in the tab 'Specs + RE'. Remember that the maximum aerodynamic efficiency can not be calculated directly but has to be solved using a numerical iteration. The Newton-Raphson method is applied using equations (4.9) and (4.10). The Newton-Raphson method is shown here below:

$$x_{n+1} = x_n - \frac{f(x_n)}{\frac{df(x_n)}{dx}} \tag{5.62}$$

With

$$f(x_n) = (4.9)$$

$$\frac{df(x_n)}{dx} = \text{(4.10)}$$

In the Excel file, there are ten iterations executed to calculate the maximum aerodynamic efficiency. The iteration converges quickly, thus it is impossible that the amount of iterations is not sufficient. The iteration is found on the bottom of this tab. The eventual value for the maximum aerodynamic efficiency is shown in the red field, represented on the picture below.

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,855	
Aspect ratio	A	7,67	
Bypass ratio	μ	4,85	
Wing loading	m_{MTO}/S_W	670	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	

Variables

Cruise matches the design point	V/V_{md}	**1,0**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,018
Lift coefficient at E_{max}	$C_{L,md}$	0,61
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	0,607
Actual aerodynamic efficiency, cruise	E	16,88
Max. glide ratio, cruise	E_{max}	**16,88**

Newton-Raphson for the maximum lift-to-drag ratio

Iterations	1	2	3
f(x)	0,10	0,00	0,00
f'(x)	-0,11	-0,12	-0,12
E_{max}	16	16,89	16,88

Figure 5.17 Screenshot: Reverse Engineering.xlsm – 2) E_max

5.5 Specific Fuel Consumption

To calculate the specific fuel consumption of the aeroplane, the user has to give up if the aeroplane is a transport jet or a business jet in the section 'Mission fuel fraction'. According to the type of jet, the fuel fractions will modify automatically. It is also possible to fill out own values for a specific mission. Besides this, the user has to give up if it concdrns a domestic flight or an international flight. According to the choice made here, the amount of reserve fuel will modify.

To calculate the specific fuel consumption, the payload mass and operating empty mass must be 'known' in the tab 'Specs + RE'. The specific fuel consumption is then calculated using equation (4.11). It is important that the payload mass matches with its range. If the maximum range is used to calculate the specific fuel consumption, there is also another way to calculate this using the available fuel volume of the aeroplane. The reason becomes clear with the payload-range diagram of a random aeroplane (Figure 5.18). It is not possible to reach the maximum range while flying with maximum payload. When intending to fly at maximum range, the entire fuel capacity is used and therefore it is possible to calculate the specific fuel consumption using the available fuel volume of the aeroplane. But this also means that it is not always the case that an airplane flies with a maximum amount of fuel. It is more benifitial to fly with more payload and less fuel (for as long the destination can be reached). Since the first law of aircraft design takes the payload and the fuel into account, this approach can be used to calculate the specific fuel consumption when the aeroplane does not fly its maximum range.

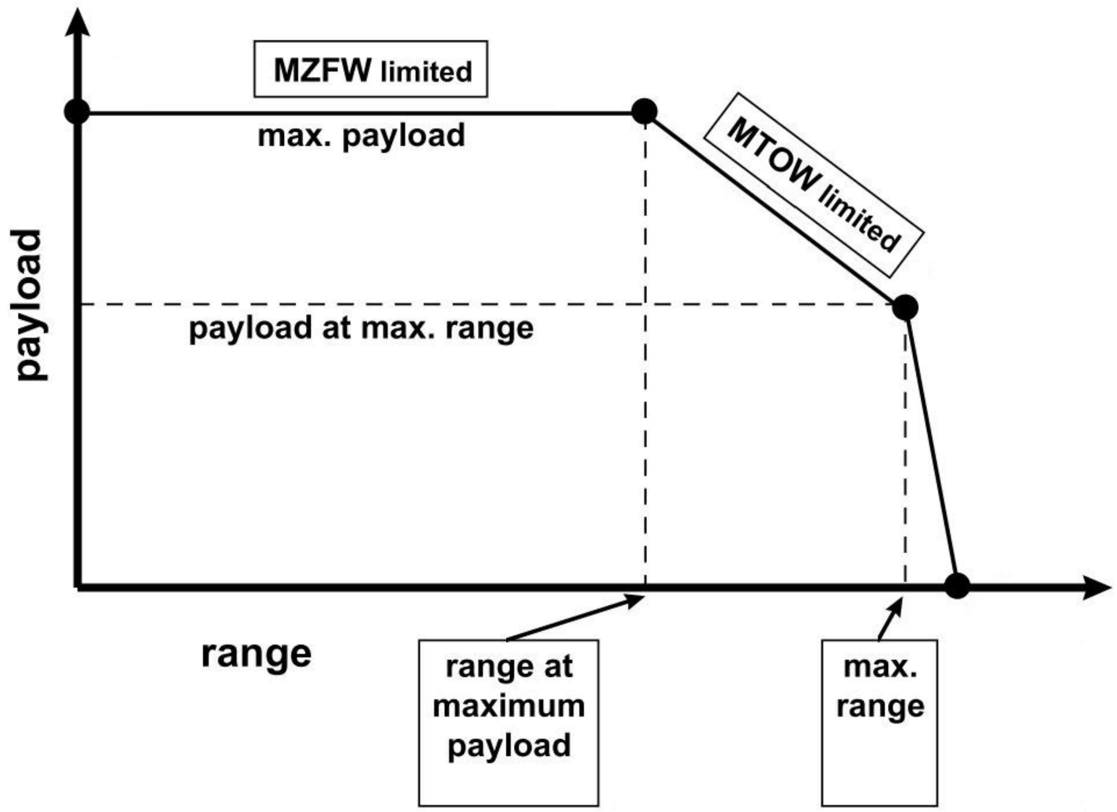


Figure 5.18 Generic payload-range diagram (based on **Scholz 2015**)

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	7500	NM
Mach number, cruise	M_{CR}	0,855	
Bypass ratio	μ	4,85	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	
Available fuel volume	$V_{fuel,available}$	204,35	m³
Maximum take-off mass	m_{MTO}	362870	kg
Mass ratio, landing - take-off	$m_{PL/}m_{MTO}$	0,050	
Mass ratio, operating empty - take-off	$m_{OE/}m_{MTO}$	0,505	

Calculated values			
Actual aerodynamic efficiency, cruise	E	16,88	
Cruise altitude	h_{CR}	11492	m
Cruise speed	V_{CR}	252	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,446	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,554	
Available fuel mass	$m_{F,available}$	163480	kg
Relative fuel mass (acc. to fuel capacity)	$m_{F,available}/m_{MTO}$	0,451	
Mission fuel fraction (acc. to fuel capacity)	M_{ff}	0,561	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**no**	
	international	**yes**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	1064900	m
Loiter time	t_{loiter}	1800	s
Specific fuel consumption (acc. to PL and OE)	SFC	**1,45E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**1,42E-05**	kg/N/s

Figure 5.19 Screenshot: Reverse Engineering.xlsm – 3) SFC

5.6 Verification

The tab '4) Verification' takes no part in the reverse engineering. It is only developed in order to perform a theoretical verification on the reverse engineering results. The intention of this enlargement of the file is only to have a verification value to compare with the reverse engineering results. In this his way, a deviation can be made and is it possible to confirm the functioning of the reverse engineering method.

To perform the verification calculations, the user has go to the tab 'Specs + RE' and fill out the data to execute the verification. The following parameters must be known:

Sweep angle	φ_{25}
Mean aerodynamic chord	c_{MAC}
Position of maximum camber	$x_{(y\_c),max}$
Camber	$(y_c)_{max}/c$
Position of maximum thickness	$x_{t,max}$
Relative thickness	t/c
Taper	λ

When the relative thickness is unknown, it is calculated by using the cruise Mach number as presented in equation (5.61).

5.6.1 Maximum Lift Coefficient for Landing and Take-off

Assuming that all the aircraft specifications are filled out by the user, in order to perform the reverse engineering calculations, it is now only necessary to give up the airfoil type[3] to calculate the maximum lift coefficient of the wing. To make the calculation for the maximum lift coefficient, it is devided into separate contributions. Equations (5.63) and (5.64) are used to calculate the maximum lift coefficient.

$$c_{L,max,clean} = (c_{L,\max})_{base} + \Delta_1 c_{L,max} + \Delta_2 c_{L,max} + \Delta_3 c_{L,max} \tag{5.63}$$

$$C_{L,max,clean} = \left(\frac{C_{L,max}}{c_{L,max}}\right) \cdot c_{L,max,clean} + \Delta C_{L,max} \tag{5.64}$$

3 If the airfoil is not available in the list (NACA-series), the user fills out an own value for the $\Delta y/(t/c)$ in the tab 'Data'.

Figure 5.20 gives an example from what the user will see on this point. It starts with a summary of the needed aircraft and wing specifications. Then it continues with the calculations for the maximum lift coefficient of a clean wing.

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	64,44	m
Structural wing span	$b_{W,struct}$	81,22	m
Wing area	S_W	541,2	m²
Aspect ratio	A	7,67	
Sweep	φ_{25}	37,5	°
Mean aerodynamic chord	c_{MAC}	9,68	m
Position of maximum camber	$x_{(y\_c),max}$	30	%c
Camber	$(y_c)_{max}/c$	6	%c
Position of maximum thickness	$x_{t,max}$	35	%c
Relative thickness	t/c	9,4	%
Taper	λ	0,275	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	75,10	m/s
Mach number, landing	M_{APP}	0,23	
Mach number, cruise	M_{CR}	0,855	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,4	%c
Leading edge sweep	φ_{LE}	41,7	°
Reynoldsnumber	Re	5,2E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,39	
Correction term, camber	$\Delta_1 c_{L,max}$	0,40	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,12	
Cerrection term, Reynolds' number	$\Delta_3 c_{L,max}$	0,010	
Maximum lift coefficient, airloil	$c_{L,max,clean}$	1,923	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,74	
Correction term, Mach number	$\Delta C_{L,max}$	-0,02	
Lift coefficient, wing	$C_{L,max}$	1,40	

Figure 5.20 Screenshot: Reverse Engineering.xlsm – 4) Verification – a) Maximum lift coefficient

Once the maximum lift coefficient for a clean wing is determined, the influence of the flaps on the lift coefficient is calculated. Therefore, the user has to know how many different types of flaps are used along the span and which types of flaps are used. The contribution of the flaps on the lift coefficient is calculated with the following formula:

$$\Delta C_{L,max,f} = \Delta c_{L,max,f} \cdot \frac{S_{W,f}}{S_W} \cdot K_{\varphi} \tag{5.65}$$

$\Delta c_{L,max,f}$ is the increase of the maximum lift coefficient of the airfoil produced by the flaps. It is calculated according to Table 5.7 as a percentage of the maximum lift coefficient of the clean wing, depending on the used type of high-lift system.

The second parameter is the flapped area $\frac{S_{W,f}}{S_W}$. In the Excel file, the user gets the choice to calculate with the flapped area or the flapped span. The flapped span is easier determined but it has the disadvantage that the result is less accurate. If the flapped span is used, the length of the flaps has to be measured along the leading edge (not perpendicular on the plane of symmetry). This is because the flapped span is calculated with the structural span.

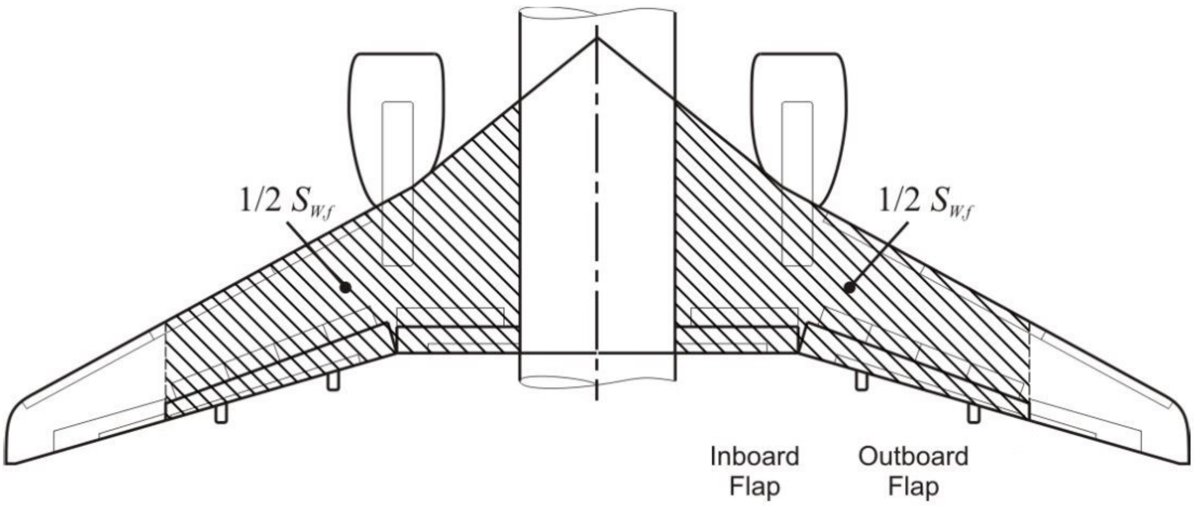


Figure 5.21 Definition of the flapped area (**Scholz 2015**)

The last parameter $K\varphi$ is a correction for the influence of the sweep. The lift coefficient decreases with increasing sweep angle. The same counts for the flaps; the effectiveness of the high-lift system decreases when the sweep angle increases. This relation is shown in **Figure 5.22** and can be calculated with the equation (5.66).

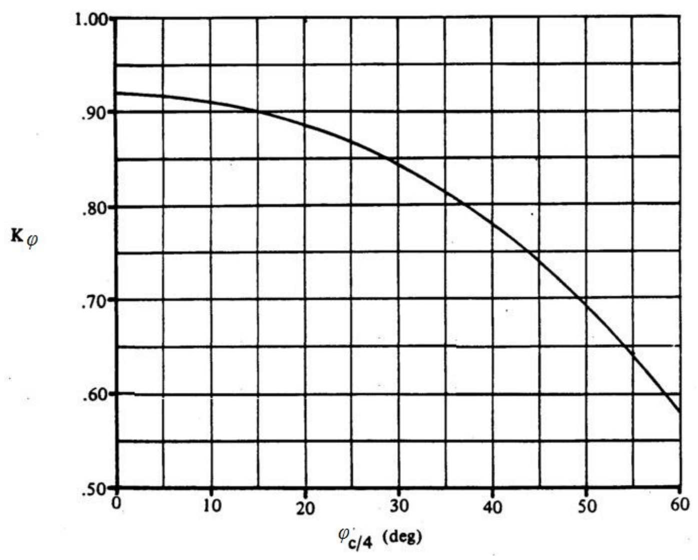


Figure 5.22 Correction factor for sweep (**DATCOM 1978**)

$$K_\varphi = \left(1 - 0{,}08 \cdot cos^2(\varphi_{c/4})\right) \cdot cos^{3/4}(\varphi_{c/4}) \tag{5.66}$$

The integration of the amount of flap type groups, the different flap types and the manner to calculate the influence of the flaps are all done using drop down menus. The following figure shows the layout.

Calculations increase of lift coefficient due to flaps		**1 flap type**
Correction factor, sweep	K_φ	0,80
• Flap group A		
Double-slotted flap	$\Delta c_{L,max,fA}$	1,37
Use flapped span	b_W,fA	**34,85** m
Percentage of flaps allong the wing		43%
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,47
• Flap group B		
0,3c Plain flap	$\Delta c_{L,max,fB}$	0,72
Use flapped span	b_W,fB	**0** m
Percentage of flaps allong the wing		0%
Increase in maximum lift coefficient, flap group B	$\Delta C_{L,max,fB}$	0,00
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,47

Figure 5.23 Screenshot: Reverse Engineering.xlsm – 4) Verification – b) Maximum lift coefficient

For the leading edge flaps, the same reasoning is followed. The influence of the slats is determined with equation (5.67).

$$\Delta C_{L,max,s} = \Delta c_{L,max,s} \cdot \frac{S_{W,s}}{S_W} \cdot \cos(\varphi_{H.L.}) \tag{5.67}$$

The increase of the lift coefficient is dependent on the type of slats (Table 5.8). In the Excel file, the user has the possibility to use two different types of slats along the leading edge. When this is the case (e.g. Boeing 747-400), the user has to make sure that the flapped area (or flapped span) belongs to the right slat group. The last parameter takes the sweep into account by using the sweep angle of the hinge line where the slats turn around. The layout and integration in Excel is done the same way as with the flaps.

Calculations increase of lift coefficient due to slats		**2 slat types**	
Sweep angle of the hinge line	$\varphi_{H.L.}$	**42**	°
• Slat group A			
0,1c Kruger flap	$\Delta c_{L,max,sA}$	0,64	
Use slatted span	b_W,sA	**15,26**	m
Percentage of slats allong the wing		19%	
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,09	
• Slat group B			
0,3c Nose flap	$\Delta c_{L,max,SB}$	0,87	
Use slatted span	b_W,sB	**42,27**	m
Percentage of slats allong the wing		52%	
Increase in maximum lift coefficient, slat group B	$\Delta C_{L,max,sB}$	0,34	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,42	

Figure 5.24 Screenshot: Reverse Engineering.xlsm – 4) Verification – c) Maximum lift coefficient

Eventually, the lift coefficient for landing can be calculated with equation (5.68). Keeping in mind that during landing, the flaps are fully deflected, there is a nose-down moment around the pitch axis. This moment is countered, using the trim which generates negative lift. To take this negative lift into account, the factor 0,95 is integrated in the formula.

$$C_{L,max,L} = C_{L,max,clean} + 0{,}95 \cdot \Delta C_{L,max,f} + \Delta C_{L,max,s} \tag{5.68}$$

In some literature, the maximum lift coefficient for take-off is 80% of the maximum lift coefficient for landing. Experience shows that this relation is not reliable. Therefore, the ratio of the maximum lift coefficient for take-off and landing from the reverse engineering results is used. This means that the deviation for both maximum lift coefficient for landing and take-off, between the reverse engineered results and the verification values, will be the same.

$$C_{L,max,TO,control} = C_{L,max,L,control} \cdot \frac{C_{L,max,TO,RE}}{C_{L,max,L,RE}} \tag{5.69}$$

Since this belongs to the '4) Verification' tab, it is a matter of course that the deviation between the reverse engineered result and verification value is determined. This is shown in Figure 5.25.

Wing		
Control value maximum lift coefficient, landing	$C_{L,max,L}$	**2,27**
RE value maximum lift coefficient, landing		2,36
Control value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,98**
RE value maximum lift coefficient, take-off		2,06
	-4%	

Figure 5.25 Screenshot: Reverse Engineering.xlsm – 4) Verification – d) Maximum lift coefficient

5.6.2 Maximum Aerodynamic Efficiency

The maximum aerodynamic efficiency is theoretically calculated, the same way it is calculated for the preliminary sizing, with equation (2.13). Nevertheless, there are some differences the way it is calculated for the verification.

The constant k_E gives a value for the relation between the aerodynamic efficiency and the relative wetted aspect ratio. In paragraph 2.5, this constant is 14,9 according to **Loftin 1980** and is 15,8 according to **Raymer 1989**. For the verification, these values are not used. Instead, the k_E will be dependent of the range. **Schlüter 2006** used the data from **Raymer 2012** and sorted the airplanes according to long, medium or short range flights. This devided Figure 2.4 into three clouds of points, resulting in Figure 5.26.

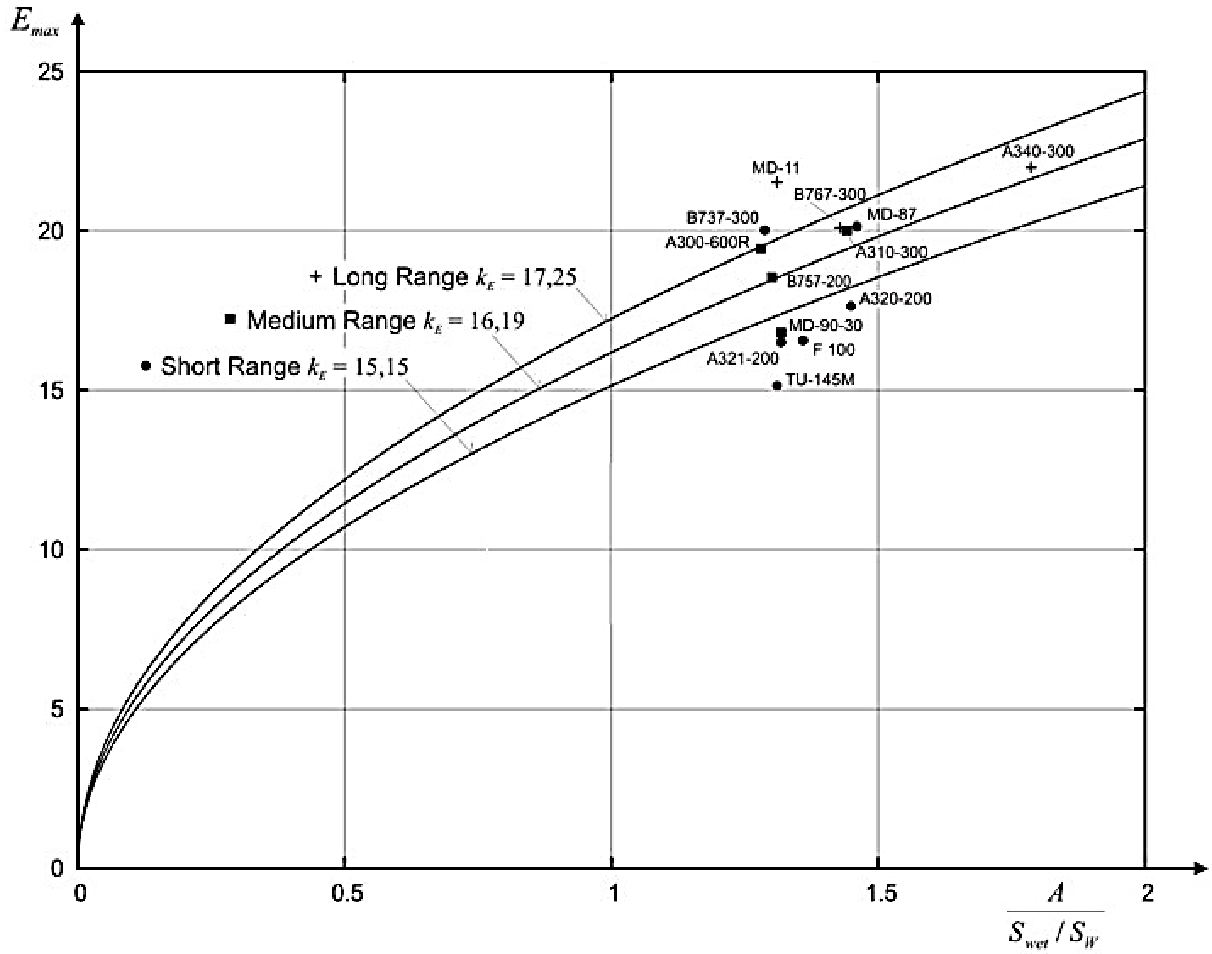


Figure 5.26 Estimation of aerodynamic efficiency, wetted area and wing area (**Scholz 2015**)

The Excel file will use these values to verify the maximum aerodynamic efficiency. The k_E is automatically adapted depending on the range of the aeroplane. Table 5.10 shows which k_E values Excel uses according to the range intervals.

Table 5.10 k_E according to the range of the aeroplane

Range		k_E
Schort range	$R \leq 3000$ NM	15,15
Medium range	3000 NM $< R <$ 5500 NM	16,19
Long range	$R \geq 5500$ NM	17,25

The '4) Verification' tab gives the user the possibility to take winglets into account. Winglets have an influence on the aspect ratio. For a simple end plate configuration, the effective aspect ratio is depending on the winglet height and the span of the aeroplane, this is shown in equation (5.70).

$$\frac{A_{eff}}{A} = k_{e,WL} = \left(1 + \frac{2}{k_{WL}} \cdot \frac{h}{b}\right)^2 \tag{5.70}$$

$$k_{WL} = 2{,}83$$

To give the user more possibilities, different non-planar configurations are integrated which are shown in Table 5.9. The effective aspect ratio is then calculated as followes:

$$\frac{A_{eff}}{A} = k_{e,WL} = k_{e,NP} \tag{5.71}$$

The last parameter is the relative wetted area. This ratio is found the same way it is for the preliminary sizing, using Figure 2.5. If the aeroplane is not mentioned on the figure, a typical value for jet powered passenger aircraft is a value between 6,0 and 6,2.

This way, the maximum aerodynamic efficiency is calculated with some small adaptions of the original formula. The k_E is dependent on the range, the effective aspect ratio is used in order to integrate the use of winglets and the relative wetted area is unchanged. This results in equation (5.72). This value is then used to verify the reverse engineering value for maximum aerodynamic efficiency:

$$E_{max} = k_E(R)\sqrt{\frac{A_{eff}}{S_{wet}/S_W}} \tag{5.72}$$

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83	
End plate	$k_{e,WL}$	1,02	
Span	b_W	64,44	m
Winglet height	h	**0,89**	m
Aspect ratio	A	7,67	
Effective aspect ratio	A_{eff}	7,82	
Efficiency factor, short range	k_E	17,25	15,1439091
Relative wetted area	S_{wet}/S_W	**6,30**	
Control value maximum aerodynamic efficiency	E_{max}	**19,2**	
RE value maximum aerodynamic efficiency		16,88	

14%

Figure 5.27 Screenshot: Reverse Engineering.xlsm – 4) Verification – Maximum aerodynamic efficiency

5.6.3 Specific Fuel Consumption

The specific fuel consumption is calculated according to **Herrmann 2010**. The theory will not be explained here. The method is integrated in the '4) Verification' tab. To calculate this, everything is already known so the user does not need to fill out more parameters. But, the overall pressure ratio OAPR and the turbine entry temperature TET are calculated with a formula for the user who does not know these engine parameters. Unfortunately, this does not always meets the reality. It is interesting for the user, in order to achieve a more reliable value for the specific fuel consumption, to do research for this data. Paragarph 5.9 Tips and Tricks explains where this data can be found. In the end, the verification value for the specific fuel consumption is compared with the value from the reverse engineering. If the maximum range is used and the fuel capacity is filled out, the user will get two separate comparisons. One for the reverse engineered value linked to the operating empty mass and payload mass. And one linked to the fuel capacity of the aeroplane.

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,855	
Cruise altitude	h_{CR}	10622	m
By Pass Ratio	μ	4,85	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	253,00	kN
Overall Pressure ratio	OAPR	**30,20**	
Turbine entry temperature	TET	**1488,38**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,95	
Ventilator efficiency	$\eta_{ventilator}$	0,88	
Compressor efficiency	$\eta_{compresor}$	0,86	
Turbine efficiency	$\eta_{turbine}$	0,90	
Nozzle efficiency	η_{nozzle}	0,99	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	219,11	K
Dimensionless turbine entry temperature	ϕ	6,79	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,15	
Temperature function	χ	1,89	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	2,21	
Control value specific fuel consumption	SFC	0,61	kg/daN/h
Control value specific fuel consumption	SFC	**1,69E-05**	kg/N/s
RE value specific fuel consumption (acc. to PL and OE)	SFC	1,45E-05	kg/N/s
	16%		
RE value specific fuel consumption (acc. to fuel capacity)	SFC	1,42E-05	kg/N/s
	19%		

Figure 5.28 Screenshot: Reverse Engineering.xlsm – 4) Verification – Specific fuel consumption

5.7 Matching Chart

In the use, visuals and plotting, nothing changed from the preliminary sizing. The only difference is which values are known and which are calculated in order to determine the points for the matching chart. This does not influence the working of the matching chart. The points which are plotted on the matching chart are located in the tab '5b) Matching Chart_points'.

The matching chart gives a quick graphical confirmation in order to verify if the inputs by the user were done correctly. When extraordinary shapes are acquiered, the user can know with a quick view that something is wrong.

Besides this, the matching chart makes it possible to compare with a matching chart from a preliminary sizing process.

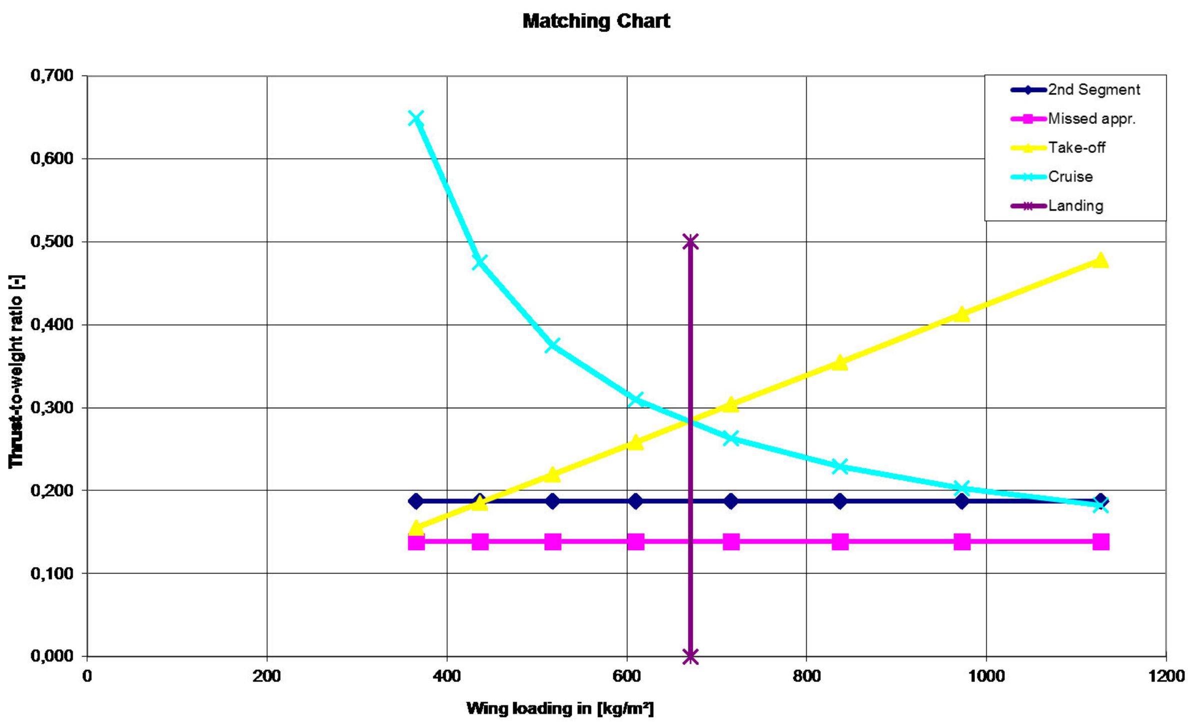


Figure 5.29 Screenshot: Reverse Engineering.xlsm – 5a) Matching Chart

5.8 Operating Instructions for 'Reverse Engineering.xlsm'

In this paragraph, the user is told how to work with the program. A brief version of the operating instructions can be found in the Excel file under the tab 'Instructions'. In order to be user friendly, the program is build-up using colour code and drop down menus. In a few cells, where it is not obvious what to do, additional information is shown when the cell is selected.

In general, the bold blue values represent input. These cells should be filled out by the user. There is no possibility one can make the program unusable by changing these values. Cells with another layout should not be touched unless the user is aware of the consequences and knows how to handle this. Blue values (not bold) are parameters based on experience. Black values are calculated interim or repeated values. The bold red values are the actual results which interest the user. The final colour is light grey, these values can be either parameters that do not apply or upper and lower limits.

5.8.1 Execute the Reverse Engineering

To start the reverse engineering, the user goes to the tab 'Specs + RE' and does the necessary research about the aeroplane that has to be reverse engineered. The aeroplane specifications need to be filled out, starting with changing the status of a few parameters to 'Known' or 'Unknown'. If the case occurs that the take-off field length or the wing span is unknown or if both the landing field length and the approach speed are unknown, the user goes to the tab 'Data – SKYbrary' where the aircraft category is filled out, using the drop down menus. Extra attention is required when the numerical classification of the category ICAO Aerodrome Reference Code equals four. When this occurs, the user has to give an upper limit for the take-off field length. The range status should also be adapted. The user gets a drop down menu with the following options for the range: range for maximum payload, range for maximum PAX (number of passengers), maximum range and the possibility to use another range according to the payload range diagram of the aeroplane. The available volume of fuel is only to be filled out when the maximum range is used. Now that every parameter has a status, the user fills out all the values.

Next is the data to optimize V/V_{md}. The actual cruise speed and cruise altitude of the aeroplane is filled out. When one of these parameters is unknown, the user has to fill out an upper and lower limit for this. If necessary, the upper and lower limits for V/V_{md} can be adapted. Initial it is set in a way that the lower limit is the minimum drag speed and the upper limit is the maximum range speed.

The next step is to choose a certification basis in the tab '1) C_Lmax' under the section 'Missed approach'. Choosing FAR Part 25 will add profile drag due to the extended landing gear. The other certification basis, JAR-25 or CS -25, does not integrate an additional drag caused by the landing gear.

As a final step, the user goes to the tab '3) SFC'. In the section 'Mission fuel fraction' there is a drop down menu for the user where one can select if the aeroplane is a transport jet or a bussines jet. According to this choice, the mission fuel fraction will be modified. Since the mission is not standard or the same for every plane, the user can adapt these values without causing any problems in the program. The last input for the user is to assign the type of flight to the aeroplane, wether it is a domestic or international flight. According to this input, the fuel reserves will modify, complying with FAR Part-121-Reserves.

Eventually, the user returns to the tab 'Specs + RE' and pushes the 'Reverse Engineering' button. The solver in Excel will start and the reverse engineering calculations are made. The results are displayed next to the button.

5.8.2 Execute the Verification

The program is initially not created to perform a verification on the reverse engineering values. It is interesting for the keen user to verify the trustworthiness of the reverse engineering calculations. The reliability of the verification values stands or falls with the accuracy of the aeroplane information.
Start on the tab 'Specs + RE' and go to the section 'Data to execute the verification'. Fill out the bold blue values. If the relative thickness is unknown, Excel will simply calculate the mean relative thickness using equation (5.61) which only depends on the cruise Mach number.

From here on, everything happens in the tab '4) Verification'. In the section 'Maximum lift coefficients' the user selects the type of airfoil. If the type is not a standard NACA profile or the user owns more detailed data, it is possible to select 'Use own type & values'. When this is the case, the user fills out the required information in the tab 'Data' section 'Airfoil'. Once this is done, the amount of flap and slat types are slected. Also the types itself are selected. Next, the user choses whether the flapped span or flapped area is used to calculate the contribution of the flaps and slats. The area gives a more accurate result but is more time consuming then using the span. When using the span, measure the length of the flaps or slats along the wing (not perpendicular to the symmetric plane). The flapped or slatted span or area is than filled out. For the slats, the sweep angle of the hinge line must be inserted. In the end, the maximum lift coefficient for landing and take-off is calculated using equations (5.68) and (5.69). Besides that, the deviation with the reverse engineering results is calculated and shown graphically directly under the verification values for the maximum lift coefficients.

In the section 'Maximum aerodynamic efficiency', the user starts by chosing the type of winglets. Note, if the winglet is an endplate, the user should also fill out the winglet height. Next, the relative wetted area must be filled out. A value is chosen using the figure beneath the calculations for the maximum aerodynamic efficiency. If the aeroplane is not on the picture, a typical value for jet powered passenger aeroplanes is a value between 6,0 and 6,2 or an own estimation can be made too. These inputs result in a verification value for the maximum aerodynamic efficiency which is compared with the value gained with the reverse engineering.

Finally, the specific fuel consumption is verified. This does not require any input from the operator. Pay attention that the overall pressure ratio (OAPR) or the turbine entry temperature (TET) can deviate a lot from the practical values. They have a big influence on the result of the specific fuel consumption. This can be an explanation if the deviation between the reverse engineered value and the verification value of the specific fuel consumption is big.

5.9 Tips and Tricks

This paragraph is made to be helpful for the user. It describes how to find a big amount of useful information in a quick and accurate way. The tips and tricks are based on own experience and provide a good basic and support for the user. To begin, several sources and platforms are mentioned and described using pros and cons concerning accuracy, reliability and integrality. In the end, an overview is shown of a comparison between the different sources.

Jane's All the World's Aircraft[4] is an annual that contains information about all the airships over the years. It is founded by John Fredrick Thomas Jane (1865 – 1916) in 1909. Since then, it is compiled and edited by many different authors. The aircraft data are detailed, complete and reliable. Because of this, its purchase is very expensive. A disadvantage is that not every airvehicle is contained in one book. The data for older aeroplanes can be found in the old editions but are left out in the new editions, unlike a dictionary. The books contain useful aircraft specifications (regarding the Excel file) such as:

Performance:	Cruising Mach number
	Take-off field length
	Landing field length
	Range
Weights and loadings:	Maximum payload
	Operating empty weight
	Maximum take-off weight
	Maximum landing weight
Dimensions:	3 view sketch

4 For this thesis; **Jane's 1954**, **Jane's 1973**, **Jane's 1982** and **Jane's 2010** are used.

Power plant:	Thrust
	Usable fuel capacity
Wing:	Wing span
Flying controls:	Leading edge devices
	Trailing edge devices
Other specifications:	Cruise speed
	Cruise altitude
	Approach speed

The second source that is worthy to consult is the **airport planning**. This is information provided by the aircraft manufacturer and can be found on their own website[5]. It gives a description about every detail from the aeroplane such as general dimensions, aircraft performance, servicing operations and maintenance preparation. The data provided by this source is integral, quite complete, reliable and for free. The only disadvantage is that the documents contain lots of unnecessary data so that it takes some time to discover the required information. Data that are needed to perform the reverse engineering and that can be found using this source are:

Performance:	Take-off field length
	Landing field length
	Payload-range diagram
Weights and loadings:	Maximum zero fuel weight
	Maximum take-off weight
	Maximum landing weight
Dimensions:	3 view drawing (detailed)
Power plant:	Engine type
	Usable fuel capacity
Other specifications:	Final approach speed

The third reference is a website on the book 'Civil Jet Aircraft Design' by L. Jenkinson, P. Simkin and D. Rhodes (**Jenkinson 2017**). The site contains more than only some details about the book, it contains aircraft industry data. This site can be used for both aircraft (**Jenkinson 2017a**) and engine (**Jenkinson 2017b**) specifications. The listing of the engine specifications are divided into three stages; take-off, climb and cruise. They are very comprehensive, accurate, user friendly and free. A few disadvantages are that some engine parameters are not expressed in SI-units and thus they need to be converted before one is able to use the values for the program. The list of different aircraft types is not large. The last disadvantage is that there is no 3 view drawing available, which makes it impossible to scale measure some parameters. The following specifications can be found with this source:

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5 Possible airport planning sources are **Airbus 2017** and **Boeing 2017**.

Aircraft specifications (**Jenkinson 2017a**)

Performance:	Cruising Mach number
	Take-off field length
	Landing field length
	Payload-range diagram
Weights and loadings:	Maximum payload
	Operating empty weight
	Maximum take-off weight
	Maximum landing weight
	Maximum zero fuel weight
	Weight ratios
Dimensions:	3 view drawing (detailed)
Power plant:	Engine type
	Number of engines
	Static thrust
	Fuel capacity (standard and optional)
	Specific fuel consumption
Wing:	Area
	Span
	Mean aerodynamic chord
	Aspect ratio
	Taper ratio
	Mean relative thickness
	25%c sweep angle
	Maximum lift coefficient for landing
	Maximum lift coefficient for take-off
High-lift devices:	Trailing edge flaps
	• Type
	• Flapped span
	Leading edge flaps
	• Type
Other specifications:	Cruise speed
	Cruise altitude
	Approach speed

Engine specifications (**Jenkinson 2017b**)

Take-off:	Thrust
	Bypass ratio
	Overall pressure ratio
	Specific fuel consumption
Climb:	Maximum thrust
Cruise:	Altitude
	Mach number
	Thrust
	Specific fuel consumption

Another interesting source that contains engine specifications is **Meier 2017**. This webpage contains data from all different kinds of jet engines. The data are not extended but it gives a good estimation. Information can be found for the specific fuel consumption, bypass ratio and the overall pressure ratio. Unfortunately, information about the turbine entry temperature (TET) is available in only a few cases.

The last interesting and free source is **SKYbrary 2017a** which contains data of 554 aeroplanes. This source provides data which are not comprehensive in comparison with the required inputs for the program. When it is consultet by the user for the first time, it is possible that the display of the information is not clear. The following, usefull information, is listed below:

Performance:	Cruising Mach number
	Take-off field length
	Landing field length
	Range
Weights and loadings:	Maximum take-off weight
Dimensions:	3 view sketch
Power plant:	Engine type
	Number of engines
	Thrust
Wing:	Span
Other specifications:	Cruise speed
	Approach speed
	Aircraft categories

The sources mentioned above are only to help the user. If all these sources are consulted and there are still a few parameters missing, that does not mean that they can not be found in another way. It is recommended to take a look on the manufactures platform. This contains lots of thrustworthy information. If by then, the user still has unknown parameters, the last option is to invoke SKYbrary[6]. This source is free and already integrated in the Excel file. A big disadvantage is that it uses intervals and thus the final value for a certain parameter depends on the solver in Excel and the accuracy from the other specifications. SKYbrary is only an option if the take-off field length, the wing span or when both the landing field length and the approach speed is unknown. Using this method is inadvisable and serves as a last possible solution to perform the reverse engineering.

[6] **SKYbrary 2017b**, **SKYbrary 2017c** and **SKYbrary 2017d**

Table 5.11 Comparison of comprehensiveness from the different sources[7]

	Jane's	Airport planning	Jenkinson 2017	Meier 2017	SKYbrary 2017 a	b/c/d
Performance						
Cruising Mach number	x		x		x	
Take-off field length	x	x	x		x	x
Landing field length	x	x	x		x	(x)
Range	x	G	x		x	
Weights and loadings						
Maximum payload	x	G	x			
Maximum operating empty weight	x	x	x			
Maximum take-off weight	x	x	x		x	
Maximum landing weight	x	x	x			
Weight ratios	(x)	(x)	x			
Power plant						
Number of engines	G	(x)	x		x	
Thrust	x		x		x	
Bypass ratio			x	x		
Fuel capacity	x	x	x			
Wing						
Area	(x)	(x)	x			
Span	x	x	x		x	x
Other specifications						
Cruise speed	x		x		x	
Cruise altitude	x		x			
Approach speed	x	x	x		x	x

[7] Directly available parameters are assigned an x-mark.
Indirectly available parameters are assigned an (x)-mark; a simple calculation with the directly known parameters is required in order to know the indirect value.
Parameters which are determined using the 3 view drawing or graphs are marked with 'G'.

Table 5.12 Quality comparison of the different sources[8]

	Jane's	Airport planning	Jenkinson 2017	Meier 2017	SKYbrary a	SKYbrary b/c/d
Comprehensiveness aircraft/engine types	5	4	2	4	5	n.a.
Comprehensiveness parameters	4	3	5	1	2	1
Accuracy	5	5	5	2	4	0
Reliability	4	5	3	2	3	0
Free	No	Yes	Yes	Yes	Yes	Yes

A scaled 3 view drawing can be very usefull in order to find some dimensions. The user can determine the wing span and area. If a verification is performed, the operator can even determine values such as the wing sweep, taper ratio and even the flapped and slatted span or area. Unfortunately measuring, scaling and calculating (especially for areas) is time consuming. If it is possible to find these parameters using a reliable source, this can be done just as well. One can also do the calculations to confirm with the source.

The user has to pay attention in measuring the wing area. Aircraft manufacturers have their own way in calculating the wing area (Figure 5.30, Figure 5.31 and Figure 5.32). Equation (5.73) shows that when the reference area changes, the lift coefficient changes in order to maintain the same value for lift. For the user, this is not a problem for the calculations, but it is a problem when the user wants to compare with lift coefficients found using different sources. When this is the case, it is important to compare the lift coefficient together with the wing area from the source.

$$L= \frac{1}{2} \cdot \rho \cdot V^2 \cdot C_L \cdot S_{ref} \tag{5.73}$$

8 Score on a scale to five, where five is the highest possible rate and zero the lowest possible score.

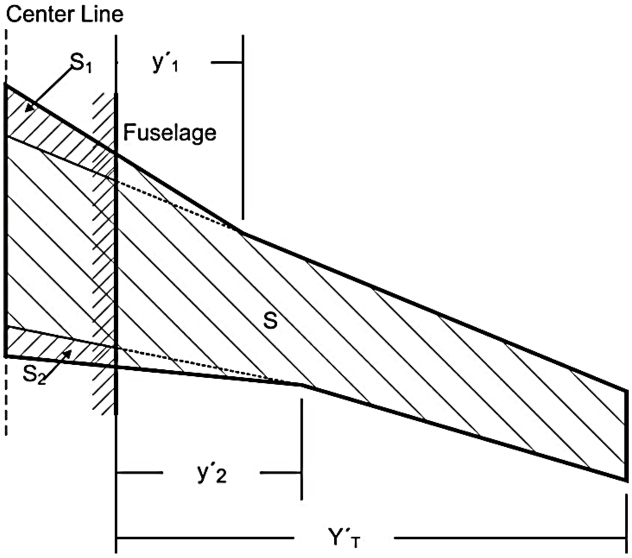

Figure 5.30 Reference area according to Boeing (**Scholz 2015**)

The aircraft manufacturer Boeing calculates the reference wing area using the formula (5.74). There is one exception, for the Boeing B-747, the definition for the reference wing area is shown with equation (5.75). These equations are supported by Figure 5.30. The definition for the reference area for the other aircraft manufacturers (Airbus, Fokker and Mc-Donnell Douglas) are obvious with Figure 5.31 and Figure 5.32.

$$S_{ref} = S + \; S_1 \cdot \frac{y_1'}{y_T'} + S_2 \cdot \frac{y_2'}{y_T'} \tag{5.74}$$

$$S_{ref} = S \tag{5.75}$$

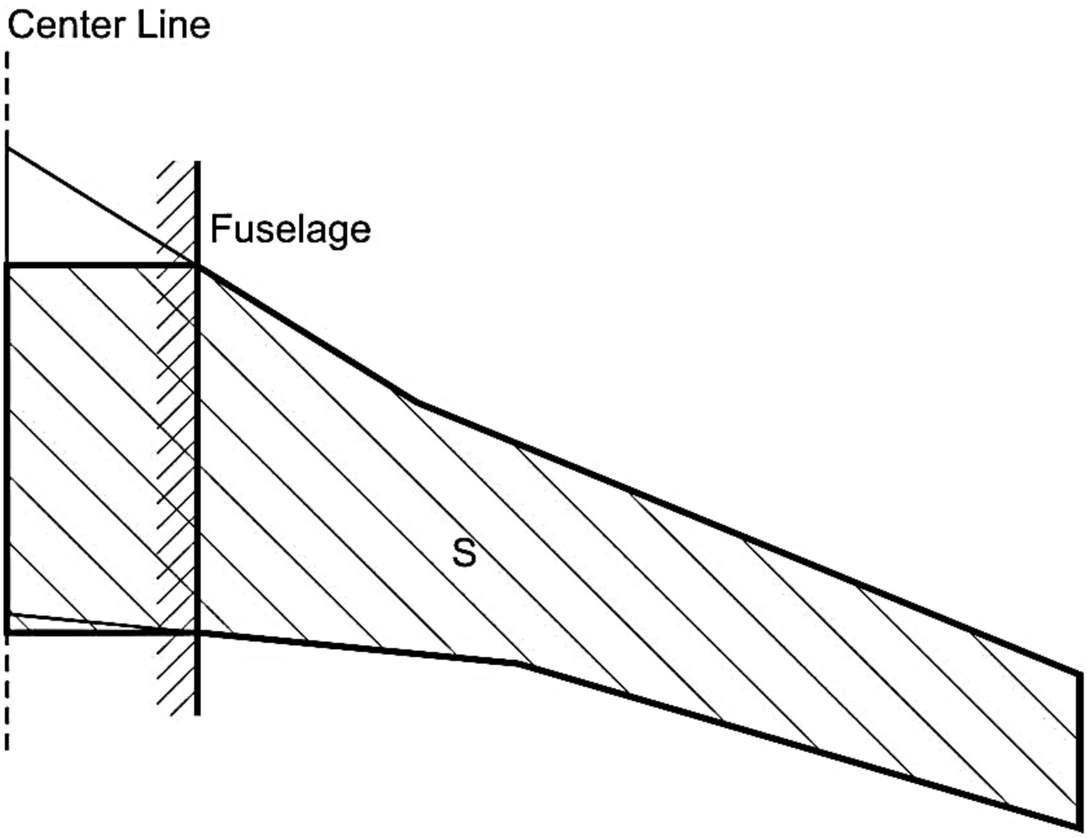

Figure 5.31 Reference area according to Airbus (**Scholz 2015**)

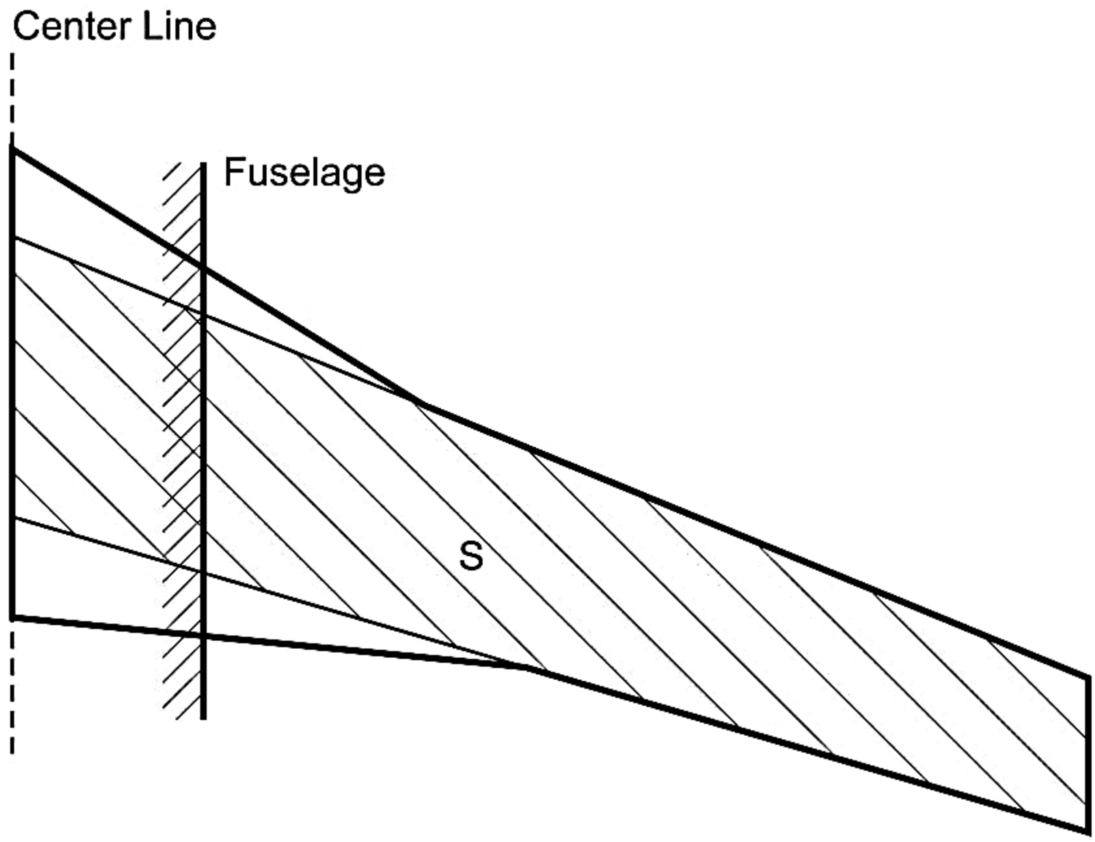

Figure 5.32 Reference area according to Fokker and Mc-Donnell Douglas (**Scholz 2015**)

6 Reverse Engineering of Passenger Jets

In this chapter, different jet powered passenger aeroplanes will be reverse engineered. Starting with one of the first passenger aircrafts the Caravelle (Sud-Aviation) and Boeing 707. Next and a little later in history, there is the BAe 146 (British Aerospace) and Boeing 747.The A320 from Airbus is also put in the program. Based on the A320, the high aspect ratio wing design 'The Rebel' and the Boeing SUGAR High are calculated. One special type of aeroplane, the blended wing body, is discussed too. The last aeroplane is the Dassault Falcon 8X business jet, a recent design based on the Falcon 7X. For each plane, data from different sources are collected, the values in the Excel file are filled out, the reverse engineering is executed and a verification calculation is made. The results are discussed and the actual Excel file for every aeroplane is added with the appendix.

For the discussion of the results, following facts are interesting to have a reference. The standard flying altitude is 35000 ft. An average relative operating empty weight of a standard jet powered aeroplane is 50%. According to **Raymer 1989**, typical values in jet transport for the wing loading is 586 kg/m² and for the thrust-to-weight ratio, it is 0,25. Since it is difficult to discuss the maximum aerodynamic efficiency, the factor for aerodynamic efficiency is calculated and compared to the values shown in Figure 5.26 and Table 5.10. The following table gives an overview of typical intervals for the maximum lift coefficient.

Table 6.1 Maximum lift coefficients for take-off and landing configuration (based on **Roskam 1989**)

Type of aircraft	$C_{L,max,TO}$	$C_{L,max,L}$
Business jet	1,6 – 2,2	1,6 – 2,6
Jet transport	1,6 – 2,2	1,8 – 2,8

Furthermore, some typical values are shown in the following table for the relative maximum landing mass. The larger the range, the smaller this ratio will be.

Table 6.2 Statistical values of the relative maximum landing mass for different types of aircraft and design range (based on **Roskam 1989** and **Loftin 1980**)

Type of aircraft	Design range (NM)	m_{ML}/m_{MTO}
Business jet		0,88
Jet transport		
• Short range	up to 1000	0,93
• Medium range	1000 – 3000	0,88
• Long range	3000 – 8000	0,78
• Ultra-long range	more than 8000	0,71

Finally, typical values for the specific fuel consumption for jets are shown below, both during cruise and during loiter.

Table 6.3 Specific fuel consumption for jets (based on **Raymer 1989**)

***SFC* [mg/N/s]**	**Cruise**	**Loiter**
Turbojet	25,5	22,7
Turbofan, low bypass ratio	22,7	19,8
Turbofan, high bypass ratio	14,2	11,3

The tool is simplified in a way that the fuel consumption for cruise and loiter are equated to each other. Since the cruise will have the largest mass fuel fraction, the specific fuel consumption for cruise will be the reference.

6.1 Caravelle (Sud-Aviation)

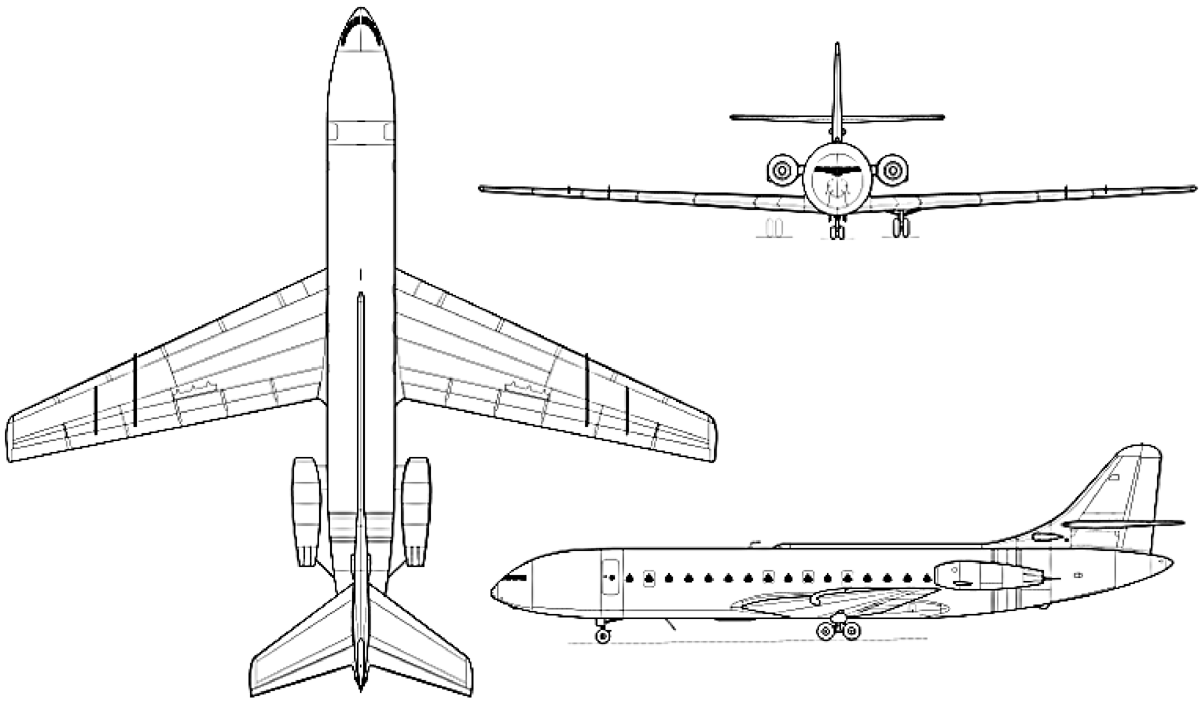

Figure 6.1 3 view drawing of the Caravelle (**Watts 2012**)

The Caravelle is a jet powered short/medium-range passenger airplane, built by the French aircraft manufacturer Sud-Aviation which first flight was in 1955. The designers chose for the rear-mounted engine configuration to reduce cabin noise. This means that the center of gravity

(CG) shifts to the back of the aeroplane and with the center of gravity, so does the wing. Because the wing is placed more to the back, the lever arm of the tail decreases. In order to counter the aerodynamic forces and ensure the controllability, stability and manoeuvrability, the tail surfaces have to be larger. This increases the operating empty weight. To avoid that the horizontal tail plane is in the exhaust stream of the engines, it has a cruciform tail. This requires a reinforced vertical tail plane to carry the horizontal tail plane. Also this increases the operating empty weight. This means that a large relative operating empty weight can be expected. The Caravelle is not a light-weight design. Because this is a very old design, the engines are also outdated. This means that it can be expected that the Caravelle has a larger fuel consumption, in comparison with the today's aeroplanes.

The airplane specifications for the Caravelle are easy to find since this is a very old aeroplane. The input values, for the tool, are taken from the Caravelle 10B. An overview of these inputs, followed by the reverse engineering results, is shown below:

Table 6.4 Input values and reverse engineering results of the Caravelle 10B

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1707	m	Large
Take-off field length	s_{TOFL}	2134	m	Large
Cruise Mach number	M_{CR}	0,765	-	Small
Design range	R	1431	NM	SR
Relative landing mass	m_{ML}/m_{MTO}	95	%	Normal
Relative operating empty mass	m_{OE}/m_{MTO}	53,7	%	Large
Wing loading	m_{MTO}/S_W	382	kg/m²	Very small
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,234	-	Small
Cruising altitude	h_{CR}	32800	ft	Small
Speed ratio	V/V_{md}	1,316	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	1,99		Normal
Maximum lift coefficient, take-off	$C_{L,max,TO}$	1,88		Normal
Maximum aerodynamic efficiency	E_{max}	17,05		Normal
Specific fuel consumption	SFC	26,8	mg/N/s	Very large

What is striking on the Caravelle is the rather low cruising altitude which is 2200 feet below the typical cruising altitude. If the Caravelle would fly higher, it could reduce the fuel consumption. But since the Caravelle does not have an oxygen system (reduce in operating empty weight), the aeroplane is forced to fly at low atlitude. In case of a sudden loss of pressurization, the pilot has to make a high-speed emergency descent of 10000 ft/min (not exceeding 0,5G). The aeroplane has to reach an altitude of usually 10000 ft in order to guarantee the passengers' safety. This is only allowed according to FAR regulations when flying at low altitude.

The maximum lift coefficient for landing is average. This is because the Caravelle has a normal relative landing mass and a quite large landing field length while the wing loading is very small. The maximum lift coefficient for take-off does not differ a lot from the lift coefficient for landing. Also this value is average. This is due to the slightly small thrust-to-weight ratio in combination with the slightly large value for the reference field length. The value for k_E is 14,9 which is a little smaller than the value for a short range aeroplane (15,5). The specific fuel consumption is very large. This is mainly caused by the small range that the aircraft can reach with its engines. These days, a larger distance can be alternated using less fuel with the same amount of thrust because of technological improvements. According to the reverse engineering tool, the optimum cruising speed is the speed to reach the maximum range (443 kt). This speed is less economical than flying at the minimum drag speed.

These reverse engineered values are compared with a theoretical verification value, which can be found in the appendix. The Caravelle has a modified NACA 65-212 as airfoil shape. According to **Jane's 1954**, the Caravelle has a single slotted flap and a fixed slat. The flapped and slatted span are measured using the 3 view drawing. Furthermore, the Caravelle has two wing fences on each wing but does not make use of winglets. The difference for the maximum lift coefficient for the reverse engineering value and the verification value is only 1% and therefore a good estimation. Also the error on the aerodynamic efficiency is small, 2%. The difference for the specific fuel consumption is just acceptable with 10%. Possible errors are that the turbine entry temperature is not correct for the verification value. Another possible shortcoming is that the mission fuel fraction of the Caravelle, deviates from the standard transport jets' mission fuel fraction (according to **Roskam 1989**).

6.2 Boeing 707

A contemporary from the Caravelle is the Boeing 707. The production of this aeroplane started in 1955 and ended in 1980. During this period, different versions of this aeroplane were developed which could carry 150 to 189 passengers. The first flight was in 1958. It is a long range jet powered aeroplane and is provided by four turbofan engines (Pratt & Whittney JT3D). The specifications of the Boeing 707-320C are standard specifications. Consequently, the values for the lift coefficient and maximum aerodynamic efficiency can be expected normal for its time. Since this is also an old aeroplane, a large specific fuel consumption is expected.

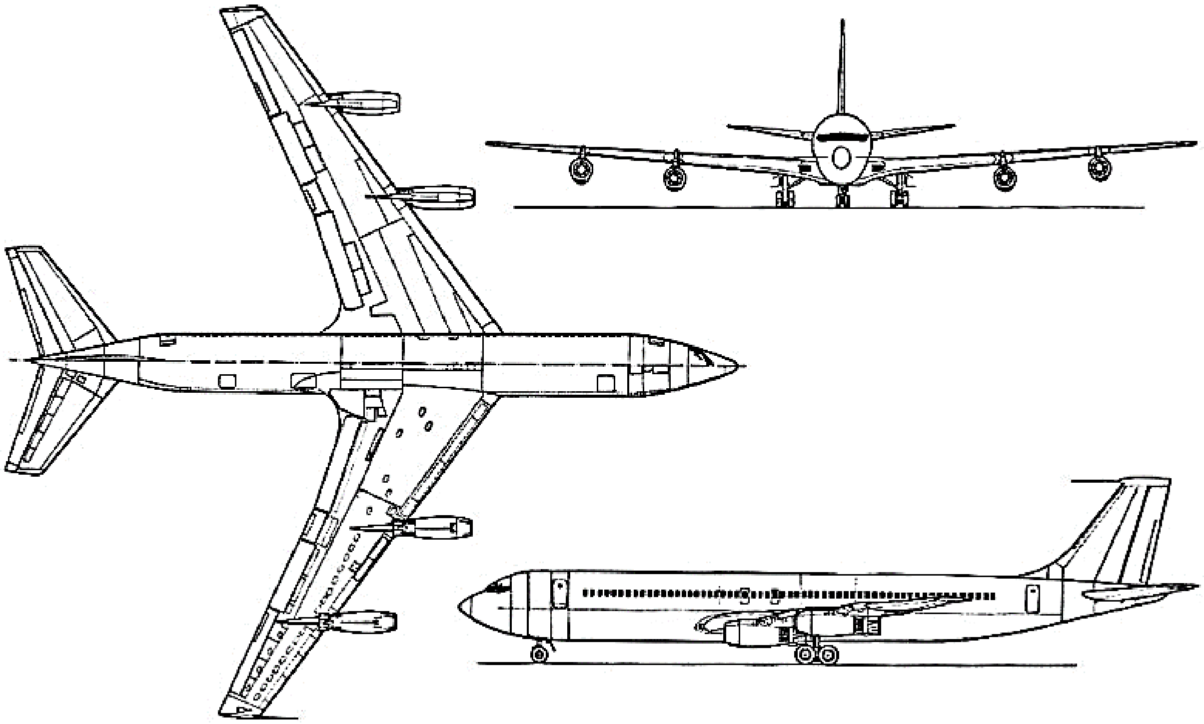

Figure 6.2 3 view drawing of the Boeing 707 (**Anderson 2017**)

Table 6.5 Input values and reverse engineering results of the Boeing 707-320C

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1905	m	Normal
Take-off field length	s_{TOFL}	3054	m	Large
Cruise Mach number	M_{CR}	0,82	-	Normal
Design range	R	5000	NM	LR
Relative landing mass	m_{ML}/m_{MTO}	74,0	%	Small
Relative operating empty mass	m_{OE}/m_{MTO}	43,8	%	Small
Wing loading	m_{MTO}/S_W	534	kg/m²	Normal
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,228	-	Small
Cruising altitude	h_{CR}	35000	ft	Normal
Speed ratio	V/V_{md}	1,110	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	1,94		Normal
Maximum lift coefficient, take-off	$C_{L,max,TO}$	1,80		Normal
Maximum aerodynamic efficiency	E_{max}	16,4		Small
Specific fuel consumption	SFC	21,7	mg/N/s	Large
Specific fuel consumption (acc. to fuel capacity)	SFC	21,0	mg/N/s	Large

The Boeing 707 has a slightly small wing loading. The landing field length, on the other hand, is slightly large. Another influencing factor is the relative landing mass, which is rather small for a long range aeroplane. The consequence of these specifications is that the aeroplane is not heavy anymore during the approach. The wings can carry a quasi-average wing loading and the landing field length is not short. This is the reason why the maximum lift coefficient for landing is average. The maximum lift coefficient for take-off is also normal. This is due to the interaction of the slightly large take-off field length and the smaller than average thrust-to-weight ratio. In that time, the engines could not provide the amount of thrust as today's engines can. This is why the thrust-to-weight ratio is rather small although it has four engines. This explains why the aeroplane needs more distance to increase the speed for take-off. If the requirements would demand for a shorter take-off field length, the designers would have to increase the lift coefficient for take-off without increasing the drag too much (e.g. by using high-lift systems). The relative operating empty weight is 44% which is smaller than the average 50%. Unlike the Caravelle, this aeroplane has wing-mounted engines and a long fuselage. Therefore, the tail can be made much smaller which reduces the structural weight. This makes the 707 a more lightweight design than the Caravelle. The k_E is equal to 15,5. Just like the Caravelle, this value is small with the only difference that the 707 is a long range aircraft. The specific fuel consumption of the Boeing 707-320C is large as expected, but quite normal for a low bypass turbofan. According to the optimization in the program, the speed ratio is 1,11. But compared to the other investigated aeroplanes, it can be concluded that the designers either choose for the maximum range speed or for the minimum drag speed. A flaw in the program is the fact that it will always take the design point to calculate the aeroplane specifications. Since the speed ratio is close to one, it can be assumed that the designers chose the minimum drag speed. When the tool executes the reverse engineering for a cruising speed equally to the minimum drag speed, the next values show up:

Maximum aerodynamic efficiency	E_{max}	17,9	
Specific fuel consumption	SFC	24,1	mg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	23,3	mg/N/s

The problem is that the cruising altitude is then 37400 ft, which deviates 7% from the original cruising altitude. The k_E is now 16,9 which deviates only a little from the reference value for long range aircrafts. The specific fuel consumption is also more realistic and deviates not a lot from the specific fuel consumption for a low bypass ratio turbofan.

Appendix B includes the reverse engineering for both the speed ratios. The Boeing 707-320C has two flap groups; double slotted flap (35% flapped span) and single-slotted fowler flap (7% flapped span). The leading edge high-lift system is a Handley Page slat which covers 77% of the structural span. Theoretically, this results in a lift coefficient of 2,0 and 1,9 for respectively landing and take-off. This deviates with 5% from the reverse engineering values.

Both (speed ratio) cases have the same maximum lift coefficients. But the maximum glide-ratio is different. The aeroplane has no winglets. This results theoretically in a maximum aerodynamic efficiency of 17,4. The reverse engineering value for the optimized speed ratio, deviates 5% from the verification value. The value for the minimum drag speed is slightly closer to the verification value and deviates with 4%. The specific fuel consumption is theoretically 24,3 mg/N/s. Comparing with the value for the optimized speed ratio, this deviates more than 9%. The verification value is almost equally to the reverse engineering value for minimum drag speed, 2% deviation.

With these results, it can be concluded that the plane flies with a cruising speed equal to the minimum drag speed. This situation strikes better with the verification value but with one remark; the cruising atltitude cannot be met as perfectly as it is with the optimized speed ratio. This only means one thing, the cruise curve on the actual matching chart does not fit the design point perfectly. It will be slightly under the design point.

Unlike the Caravelle, the Boeing 707 has an oxygen system on board in case of sudden pressurization loss. The aeroplane has four JT3D engines which are most efficient between a flight level of 33000 ft (FL330) and FL350. Above this flight level, the engines lose on efficiency which will increase the fuel consumption. The Caravelle flies at lower altitude. This is one reason why the SFC of the Caravelle is larger than the SFC of the Boeing 707.

6.3 BAe 146 (British Aerospace)

The BAe 146 is a short range aircraft which was introduced in 1982 by the British Aerospace. It is developed and manufactured in the United Kingdom. The BAe 146 is designed for short-field operations. It is powered by four turbofan engines, more specific Avca Lycoming ALF 502R-3. Remarkable at this aeroplane is the high-wing with a small anhedral of three degrees. The high-wing configuration provides the aeroplane with a good stability around the longitudinal axis and gives a good visibility for the pilots. It also makes loading easier and therefore the turnaround time can be reduced. This is essential because it is a short range aeroplane. The shorter the turnaround time, the more flights it can make in one day and thus the more profit airlines can make. With an after swept high-wing configuration, too much stability is obtained so it has to be counteracted. To cancel out some of the stabilizing momentum, anhedral is applied. Because the wing is placed on top of the aeroplane, the landing gear is embedded in the fuselage (which has to be reinforced). When the landing gear is retracted, it is encapsulated by fairings. This deforms the cylindrical form of the fuselage and causes additional drag. The BAe 146 has a short fuselage. Therefore, it needs a large tail. The aeroplane has a T-tail, this is necessary so the horizontal tail plane is out of the exhaust stream of the jet engines. This configuration has a large impact on the operating empty weight because the vertical tail plane has to be reinforced in order to carry the horizontal tail plane. However, it has the advantage that the horizontal tail plane acts as a winglet for the vertical tail plane and therefore can be made smaller, thus reducing the operating empty weight again. Conclusion; the high-wing, landing gear and large tail add extra structural weight. By analysing the 3 view drawing of the BAe 146, a large operating empty weight can be predicted. The configuration of this aeroplane makes the design heavy.

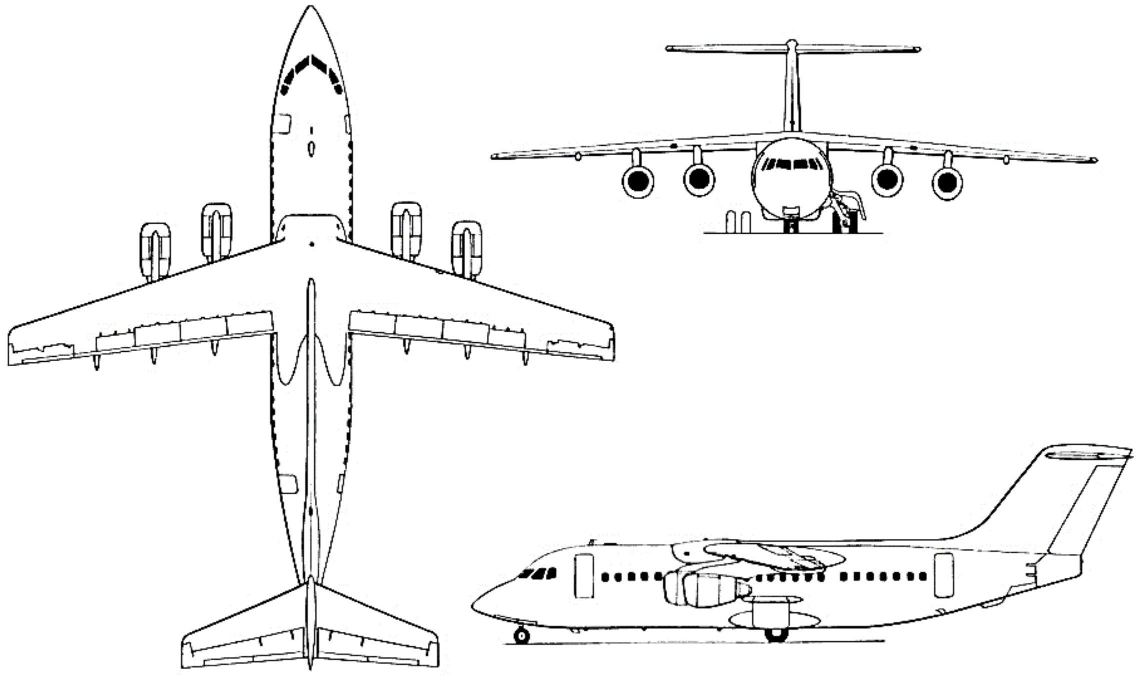

Figure 6.3 3 view drawing of the BAe 146 (**Center 2017**)

For the reverse engineering calculations, the specifications for the BAe 146-200 are used. The results for the maximum lift coefficient, maximum aerodynamic efficiency and the specific fuel consumption after applying reverse engineering are the following:

Table 6.6 Input values and reverse engineering results of the BAe 146-200

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1173	m	Very small
Take-off field length	s_{TOFL}	1564	m	Very small
Cruise Mach number	M_{CR}	0,73	-	Small
Design range	R	1000	NM	SR
Relative landing mass	m_{ML}/m_{MTO}	86,6	%	Small
Relative operating empty mass	m_{OE}/m_{MTO}	54,2	%	Large
Wing loading	m_{MTO}/S_W	525	kg/m²	Small
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,313	-	Very large
Cruising altitude	h_{CR}	30000	ft	Small
Speed ratio	V/V_{md}	1,316	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	3,62		Very large
Maximum lift coefficient, take-off	$C_{L,max,TO}$	2,63		Large
Maximum aerodynamic efficiency	E_{max}	14,5		Low
Specific fuel consumption	SFC	19,5	mg/N/s	Large

The BAe 146 has a lower than average cruising altitude of 30000 ft. The aeroplane is certified by EASA for a maximum operating altitude of FL310. This is because the pressurization system of the aeroplane comes from BAe Dynamics Analog Systems. This can be used for altitudes up to FL310 according to FAR.

The relative landing mass is small for a short range aircraft and the wing loading is slightly lower than average. Knowing this and the fact that the landing field length is very small, a very high maximum lift coefficient is needed for landing. This value is unusually high but is necessary in order to meet the requirements of the aeroplane. The BAe 146 has only trailing edge high-lift systems. It is impossible to achieve a lift coefficient that big using only flaps. But still, the short landing field length can be achieved and this by using a tail brake. Without the tail brake, the landing field length would be much larger. Therefore, it can be concluded that the actual maximum lift coefficient for landing will be smaller than calculated here. For this aeroplane, the take-off field length is rather small. This explains why the aeroplane has four engines so that there is enough thrust available to accelerate the total mass fast enough. The thrust-to-weight ratio is therefore remarkably high. These specifications result in a large maximum lift coefficient for take-off. If the aeroplane would have less engines and an average for the thrust-to-weight ratio, the lift coefficient would have to be 3,13. This maximum lift coefficient can not be reached with this wing design in take-off configuration. The maximum aerodynamic efficiency seems to be very low. This is confirmed with the k_E factor which is only 12,0. The bad maximum aerodynamic efficiency is compensated with the four engines.

Specifications about the wing and high-lift systems of the BAe 146 are hard to find. Therefore, the verification values for the maximum lift coefficient are not reliable. An attempt to estimate these values theoretically, gives 3,10 and 2,15 for respectively landing and take-off. These values are 14% smaller than the reverse engineering values, but keep in mind that the actual maximum lift coefficient for landing will be smaller because of the tail brake. Therefore, the final error with the verification value will be smaller. The maximum aerodynamic efficiency deviates with 27% which is way too much. Also the specific fuel consumption deviates too much (31%). The verification value of the specific fuel consumption looks rather large for a high bypass ratio turbofan engine.

It can be concluded that the BAe 146 is a design, based on exceptional requirements (short-field operations). The design is extraordinary in a way that it cannot be approached with theoretical standard formulas. For this reason, the theoretical verification values are unreliable in this case. According to **Jenkinson 2017a**, the maximum lift coefficient for landing is 3,43 for the Avro RJ85. This deviates only 5% from the reverse engineering values. When the tail brake could be taken into account, this error would be even smaller. According to **Meier 2017** and **Jenkinson 2017b**, the (dry) specific fuel consumption of the ALF 502R-3 for take-off is 11,6 mg/N/s. **Jenkinson 2017b** gives a specific fuel consumption during cruise of 20,4 mg/N/s (cruising Mach number is 0,7). The mutual relation between the SFC and the Mach number is linear. With some basic mathematics, it can be calculated that the SFC in the case of the BAe should be 20,7 mg/N/s. This deviates only 6% with the reverse engineering value. Other deviations from the actual aeroplane characteristics are due to the fact that the requirements for take-off demand more than the cruise in the matching chart. This means that the cruise curve will be under the design point, which is impossible for the program to calculate.

6.4 A320 (Airbus)

The A320 is a medium range aircraft, powered by two high bypass ratio turbofan engines (CFM56-5A1). It is designed and built by the aircraft manufacturer Airbus. It had its first flight in 1987. The aeroplane can carry 180 passengers and is used by many airliners. Since 2010, Airbus works on an improved version, the A320-neo (New Engine Option). This version has an improved, more economical engine and has new winglets. The 3 view drawing of the A320 shows a conventional, tail-aft aeroplane.

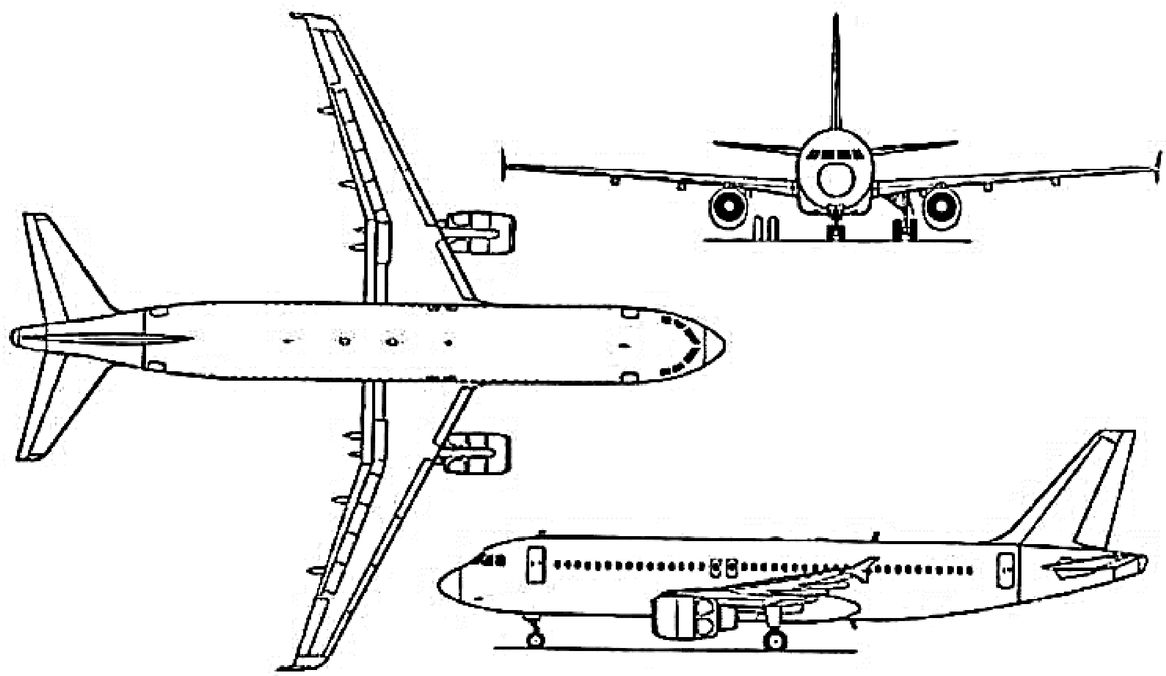

Figure 6.4 3 view drawing of the A320 (**SKYbrary 2017a**)

The reverse engineering is executed for an A320-200 with a maximum take-off mass of 73500 kg. This conventional aeroplane does not have any extraordinary requirements, thus typical values can be expected as result.

Table 6.7 Input values and reverse engineering results of the A320-200

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1700	m	Normal
Take-off field length	s_{TOFL}	2200	m	Small
Cruise Mach number	M_{CR}	0,78	-	Normal
Design range	R	1600	NM	SR
Relative landing mass	m_{ML}/m_{MTO}	87,8	%	Normal
Relative operating empty mass	m_{OE}/m_{MTO}	56,2	%	Large
Wing loading	m_{MTO}/S_W	601	kg/m²	Large
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,308	-	Very large
Cruising altitude	h_{CR}	37000	ft	Large
Speed ratio	V/V_{md}	1	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	2,90		Large
Maximum lift coefficient, take-off	$C_{L,max,TO}$	2,07		Normal
Maximum aerodynamic efficiency	E_{max}	17,9		Small
Specific fuel consumption	SFC	16,2	mg/N/s	Normal

The maximum lift coefficient will be slightly large and exceeds the standard upper limit of 2,8. This is because the landing field length and relative maximum landing mass are normal, while the wing loading is slightly large. The maximum lift coefficient for take-off is normal. This is because the large thrust-to-weight ratio is combined with a rather small take-off field length for this aeroplane. This means that the aeroplane is able to accelerate fast enough so that a larger lift coefficient is not necessary. The aerodynamic efficiency factor is 13,9 and rather small, so is the maximum aerodynamic efficiency. The specific fuel consumption is 16,2 mg/N/s and with this value, it is quite close to the reference value for a high bypass ratio turbofan engine (14,2 mg/N/s).

Unfortunately, the wing characteristics are classified, as it is for most aeroplanes. Therefore, the verification value for the maximum lift coefficient is not reliable, but an attempt is made. The A320 has a single-slotted flap and occurs along 62% of the structural wing span. The leading edge high-lift system is a nose flap and covers 82% of the structural wing span. This results in a maximum lift of 2,52 and 1,80 for respectively landing and take-off. This deviates too much from the reverse engineering values (13%). The A320 has also winglets from 2,7 m high. This has an impact on the maximum aerodynamic efficiency and thus on the specific fuel consumption. The verification value for the maximum aerodynamic efficiency is 19,6 which is 9% higher than the reverse engineering value. The verification value for the specific fuel consumption is 16,3 mg/N/s. This is 1% larger than the SFC calculated with the reverse engineering. According to **Jenkinson 2017b**, and using the same mathematics as with the Bae 146, a fuel consumption of 16,7 is determined. This deviates 3% with the reverse engineering result for the SFC.

It can be concluded that the A320 is a typical conventional aeroplane. Almost every specification of this aeroplane is average. The results for the reverse engineering are acceptable. Only the maximum lift coefficient can not be confirmed properly.

6.5 The Rebel (based on A320)

The Rebel is a design based on the A320 from Airbus. The mutual differences are shown in the table below. Remarkable is the very large value for the aspect ratio. It has increased with 267%. This is due to the enlargement of the wing span and the reduction of the wing area. Because the wing surface has a low value, the wing loading is very large and increased with 63% in comparison with the A320. The Rebel also makes use of a different very high bypass turbofan engine. The bypass ratio is now 15,5 which is 158% larger than the CFM56-5A1 engines of the A320. The Rebel intends to fly low and slow.

Table 6.8 Deviation from the Rebel of the A320

Quantity name	Symbol	Value A320	Value Rebel	SI Unit	Deviation
Cruise Mach number	M_{CR}	0,76	0,55	-	-28%
Wing surface	S_W	123	68	m²	-45%
Wing span	b_W	34,1	48,5	m	+42%
Aspect ratio	A	9,48	34,80	-	+267%
MTOM	m_{MTO}	73500	66000	kg	-10%
Relative payload mass	m_{PL}/m_{MTO}	0,262	0,292	-	+11%
Relative maximum landing mass	m_{ML}/m_{MTO}	0,878	0,920	-	+5%
Relative operating empty mass	m_{OE}/m_{MTO}	0,57	0,59	-	+4%
Wing loading	m_{MTO}/S_W	600	976	kg/m²	+63%
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\,g)$	0,308	0,275		-11%
Bypass ratio	μ	6	15,5	-	+158%
Cruise altitude	h_{CR}	37000	30000	ft	-19%

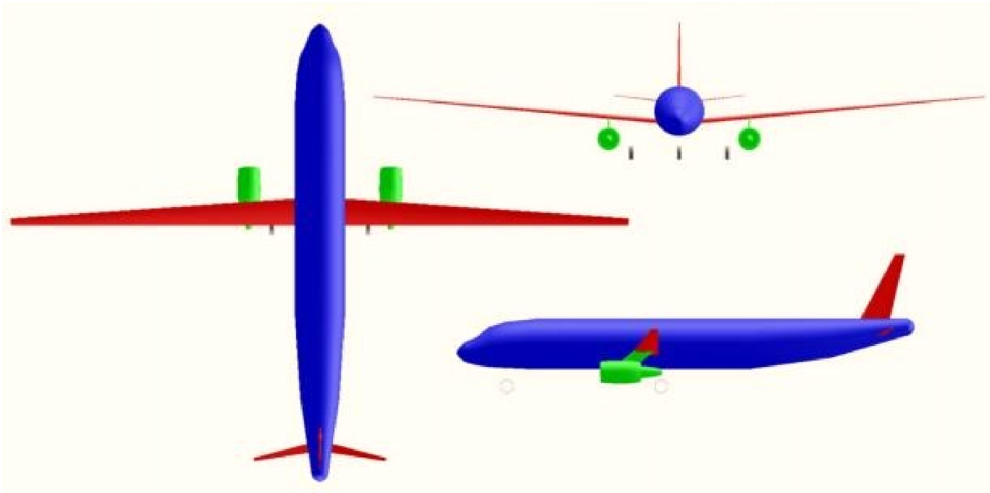

Figure 6.5 3 view drawing of the Rebel (**Johanning 2014**)

The 3 view drawing shows the changes that are made on the wing and tail. Their surfaces reduced drastically. The span is clearly larger than the wing span of the A320. The Rebel belongs to aircraft category D (according to the ICAO Aerodrome Reference Code) while the A320 still belongs to category C. This means that the number of airports, where the Rebel will be accepted, decreases.

The inputs for the reverse engineering are based on **Johanning 2014**. The Oswald efficiency factor will have a smaller value for take-off, cruise and landing because of the wing design (exact values can be found in the appendix). Just like the A320, the speed ratio is one and is therefore forced to fit. The following results are obtained when executing the program:

Table 6.9 Input values and reverse engineering results of the Rebel

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	2700	m	Large
Take-off field length	s_{TOFL}	2700	m	Large
Cruise Mach number	M_{CR}	0,55	-	Very small
Design range	R	1510	NM	MR
Relative landing mass	m_{ML}/m_{MTO}	92,0	%	Large
Relative operating empty mass	m_{OE}/m_{MTO}	59,4	%	Large
Wing loading	m_{MTO}/S_W	976	kg/m²	Very large
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,275	-	Large
Cruising altitude	h_{CR}	30000	ft	Very small
Speed ratio	V/V_{md}	1	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	3,11		Large
Maximum lift coefficient, take-off	$C_{L,max,TO}$	3,07		Very large
Maximum aerodynamic efficiency	E_{max}	24,7		Large
Specific fuel consumption	SFC	10,3	mg/N/s	Small

The maximum lift coefficient for landing is large although the landing field length is large too. This large value is only for a small part due to the large relative landing mass. It is mainly due to the very large wing loading and the small surface of the wing. The maximum lift coefficient for take-off is very large. The take-off field length is quite large and the thrust-to-weight ratio is above average. This should keep the lift coefficient low, but this is not the case. If the design of the Rebel, requires a typical value for the maximum lift coefficient for take-off, either the thrust-to-weight ratio should increase and/or the take-off field length should increase. The wing design makes it also difficult to integrate complex high-lift systems. The maximum aerodynamic efficiency is larger than it usually is for a typical aircraft. The aerodynamic efficiency factor is 12,7 which is too small according to the typical value for medium range jet transport. This is due to the small value for the Oswald efficiency factor during cruise. The specific fuel consumption of the Rebel is small. This is because the aerodynamic efficiency has a large value and the aeroplane flies at the minimum drag speed, which is the speed where the aeroplane will consume the least fuel.

Unfortunately, there is no data available on the wing except for the span and aspect ratio. This makes it impossible to determine a verification value for the maximum lift coefficients. The maximum aerodynamic efficiency can be theoretically calculated. The verification value is 29,6 which deviates 20% with the reverse engineering value. This is determined, using the aerodynamic efficiency factor from Table 5.10. This value assumes a standard Oswald efficiency factor. Since this is not the case, it is calculated with equation (2.14). This results in a maximum aerodynamic efficiency of 26,1 which only deviates 5% with the reverse engineering value. The result will not get better because the program calculates the maximum aerodynamic efficiency using the design point. The actual cruise curve on the matching chart will be under the design point and this causes the error. The specific fuel consumption according to **Herrmann 2010** is 10,3 mg/N/s. This deviation between the verification value and reverse engineered value is very small. The answer for these corresponding values lays with the source itself, **Johanning 2014**. A preliminary sizing was done for the Rebel. In order to do this, the specific fuel consumption has to be known. Therefore, **Herrmann 2010** was used to estimate the SFC for an engine with a bypass ratio of 15,5 flying at FL300 with 0,55 Mach. The reverse engineering program is based on the preliminary sizing. The correct working of the program is confirmed by the fact that the SFC is the same in this case.

The Rebel is an interesting concept and may be interesting in the future for environmental reasons. In comparison with the A320-200, 36% if fuel is saved.

6.6 SUGAR High (Boeing)

This aeroplane is based on a NASA-commissioned project of Boeing. It is one of the many studied aeroplanes in the Subsonic Ultra Green Aircraft Research (SUGAR). The first aeroplane of this project, also the reference aircraft, is the SUGAR Free. This is a conventional aeroplane with two CFM56 engines (physical properties, aerodynamics and performance are similar to the A320). The next aeroplane is the Refined SUGAR which has very high bypass turbofan engines with 2030 technologies. The third aeroplane is the SUGAR High, this aeroplane has a braced high wing and very high bypass ratio turbofan engines. The fourth aeroplane of the project is the SUGAR Volt. This is the hybrid version of the SUGAR High by making use of battery cells in association with fuel cells. The last aeroplane of this project is the SUGAR Ray, this is the flying wing version of the SUGAR High. The goal of this project is to studie different structures of an aeroplane in order to meet with the NASA fuel burn goals.

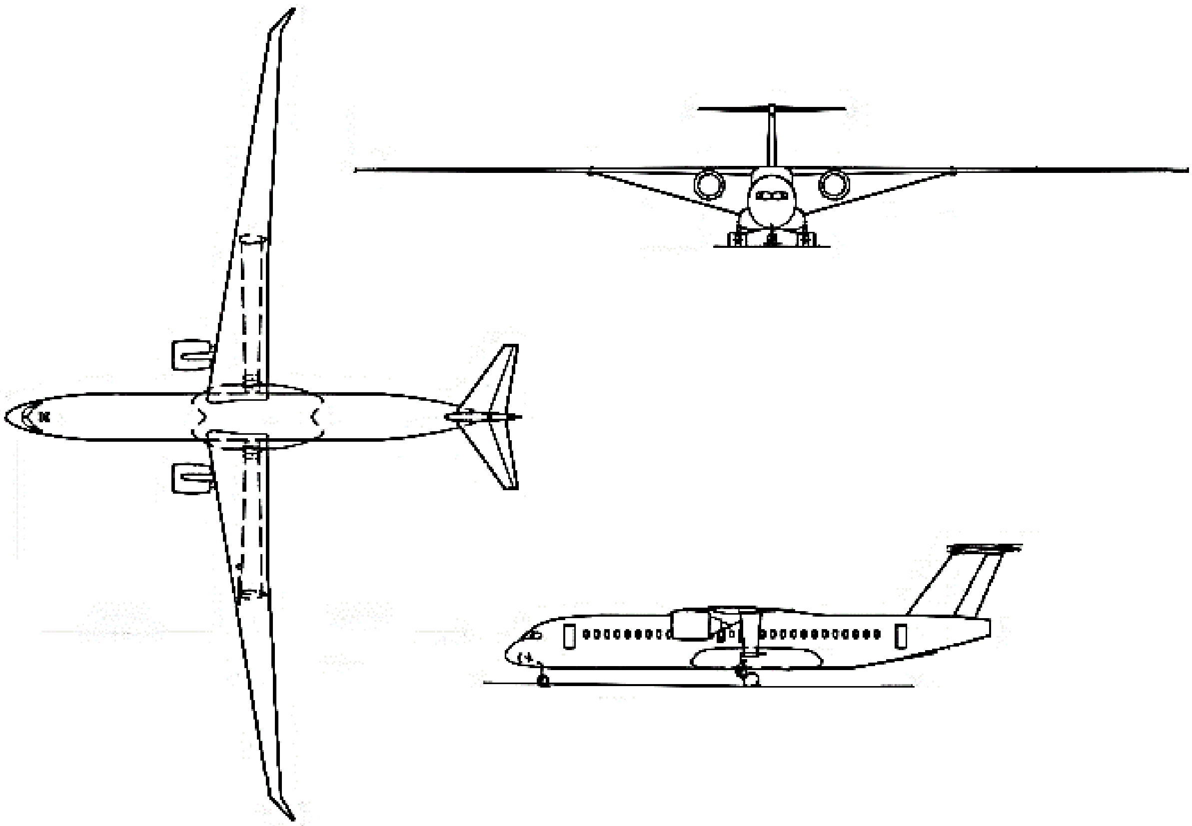

Figure 6.6 3 view drawing of the Boeing SUGAR High (**Boeing 2010**)

Table 6.10 Input values and reverse engineering results of the Boeing SUGAR High

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1056	m	Small
Take-off field length	s_{TOFL}	2496	m	Normal
Cruise Mach number	M_{CR}	0,74	-	Small
Design range	R	3500	NM	LR
Relative landing mass	m_{ML}/m_{MTO}	94,6	%	Very large
Relative operating empty mass	m_{OE}/m_{MTO}	65,2	%	Very large
Wing loading	m_{MTO}/S_W	471	kg/m²	Small
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,231	-	Normal
Cruising altitude	h_{CR}	44000	ft	Very large
Speed ratio	V/V_{md}	1,316	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	3,44		Large
Maximum lift coefficient, take-off	$C_{L,max,TO}$	1,91		Normal
Maximum aerodynamic efficiency	E_{max}	30,1		Very large
Specific fuel consumption	SFC	6,82	mg/N/s	Very small

It is remarkable that the relative landing mass is very large for a long range aeroplane. The SUGAR High is supposed to reduce the fuel consumption drastically. According to Boeing, the SUGAR High should safe fuel burn with 58%. This results in a smaller amount of necessary fuel and larger payload possibilities. This way, the SUGAR High achieves a relative landing mass for a short range aircraft. A braced wing can decrease the wing mass with 30%. This means that the relative operating empty mass should be smaller than the 50% of a conventional aircraft which makes it a light-weight design. But instead, the relative operating empty weight is very large. This is because the wing is enlarged in size. The span is almost dubble of the span of an A320 and the surface is a little higher. This larger wing reference area explains the small wing loading. When this is combined with the small landing field length, a large maximum lift coefficient for landing is required. The maximum lift coefficient for take-off is normal because of the normal values for the take-off field length and the thrust-to-weight ratio. It is known that the designers chose for the maximum range speed. Therefore, the speed ratio in the reverse engineering program is forced to fit 1,316. Doing so, this results in a maximum aerodynamic efficiency of 30,1. This value is very large because the aspect ratio of the Boeing SUGAR High is extremely large (24). This large maximum aerodynamic efficiency results in a very small specific fuel consumption. Also the high cruising altitude reduces this SFC.

Unfortunatly, the Boeing SUGAR High is only a case study. The high-lift systems and wing parameters are all unknown. Therefore, it is impossible to verify the reverse engineering results for the maximum lift coefficients. The maximum aerodynamic efficiency can be determined theoretically. The verification value is 32,1 which deviates 7% with the reverse engineering value. According to **Boeing 2010** the cruising aerodynamic efficiency is 25,97. The reverse engineering value is 26,10. The deviation with this reference is 0,5%. The verification value for the SFC is 15,2 mg/N/s which deviates 123% from the reverse engineering value. This is a value does not take the futuristic features of the engine into account. According to **Boeing 2010**, the Boeing SUGAR High should safe fuel with 58%. Assume that **Herrmann 2010** calculated the normal SFC for the Boeing SUGAR Free. Saving 58% percent of fuel equals to a necessary fuel amount of 42% of the reference aeroplane. This results in an SFC of 6,38 mg/N/s which deviates 7% with the value according to the reverse engineering.

The Boeing SUGAR High would be a great improvement in civil aviation. The large aspect ratio results in a large aerodynamic efficiency which reduces the fuel consumption of the aeroplane. An important element that is essential for fuel saving is the new engine technology. According to **Boeing 2010**, the SUGAR Refined would save 50% fuel using only the futuristic engines.

6.7 Boeing 747

The Boeing 747 (a.k.a. the Jumbo Jet) is a long range aircraft with four high bypass turbofan engines. The first flight of this aeroplane was in 1969. One year later, the aeroplane got in service. The production and use of the Boeing 747 still exist on this day. The former president and entrepreneur of Pan American World Airways (Pan Am), Juan Trippe, asked the aircraft manufacturer Boeing to design and built an aeroplane that was more than twice the size of the Boeing 707. This was the start of the first widebody. The aircraft can carry two and a half times more passengers than the Boeing 707. It has two decks, but the length of the upper deck is only a small part of the lower deck. Therefore, it is not a full dubbel deck aeroplane (like the A380 by Airbus). The aeroplane is a typical tail-aft aircraft.

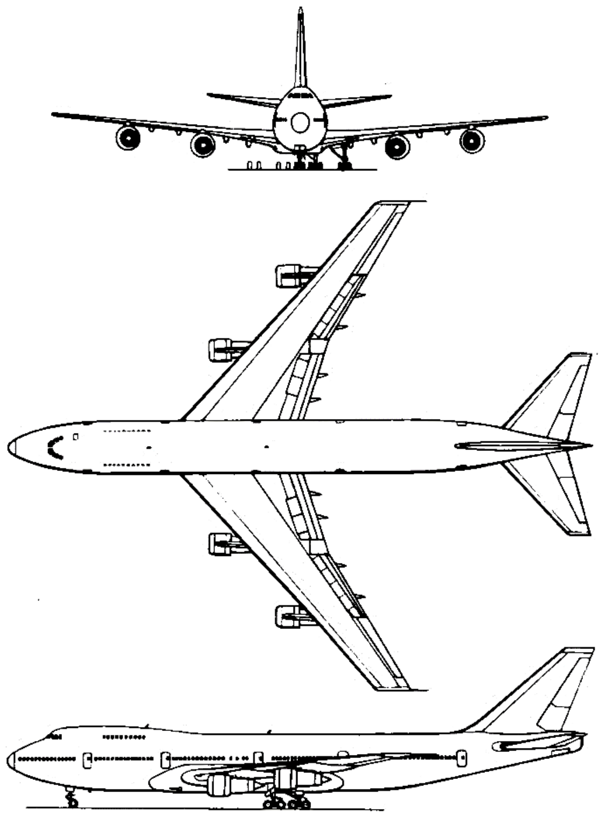

Figure 6.7 3 view drawing of the Boeing 747 (**Center 2017**)

The calculations are done for the Boeing 747-400 with the PW4056 engines. This is an imporved version of the original Boeing 747. The most important changes are the increased range, winglets and additional fuel tanks in the tail. The Boeing 747-400 entered service in 1989.

Table 6.11 Input values and reverse engineering results of the Boeing 747-400

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	1905	m	Small
Take-off field length	s_{TOFL}	2815	m	Normal
Cruise Mach number	M_{CR}	0,855	-	Large
Design range	R	4890	NM	LR
Relative landing mass	m_{ML}/m_{MTO}	71,8	%	Small
Relative operating empty mass	m_{OE}/m_{MTO}	50,5	%	Normal
Wing loading	m_{MTO}/S_W	671	kg/m²	Very Large
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\,g)$	0,284	-	Large
Cruising altitude	h_{CR}	34800	ft	Normal
Speed ratio	V/V_{md}	1	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	2,36		Normal
Maximum lift coefficient, take-off	$C_{L,max,TO}$	2,06		Normal
Maximum aerodynamic efficiency	E_{max}	16,9		Small
Specific fuel consumption	SFC	17,4	mg/N/s	Normal

The wing loading of this aeroplane is extremely large. This is caused directly by the size of the aeroplane and can be explained by the square cube law. If an object grows in size than the volume will increase faster than the surface does. The same law, translated using factors; if an object doubles in size (2x), the surface of the object will quadruple (4x) and the volume will octuple (8x). Thus, if an aircraft with average specifications, increases in size, the mass will increase with a factor eight and the surface increases with a factor four. If an average wing loading has to be achieved, the wing surface has to increase (Figure 6.8).

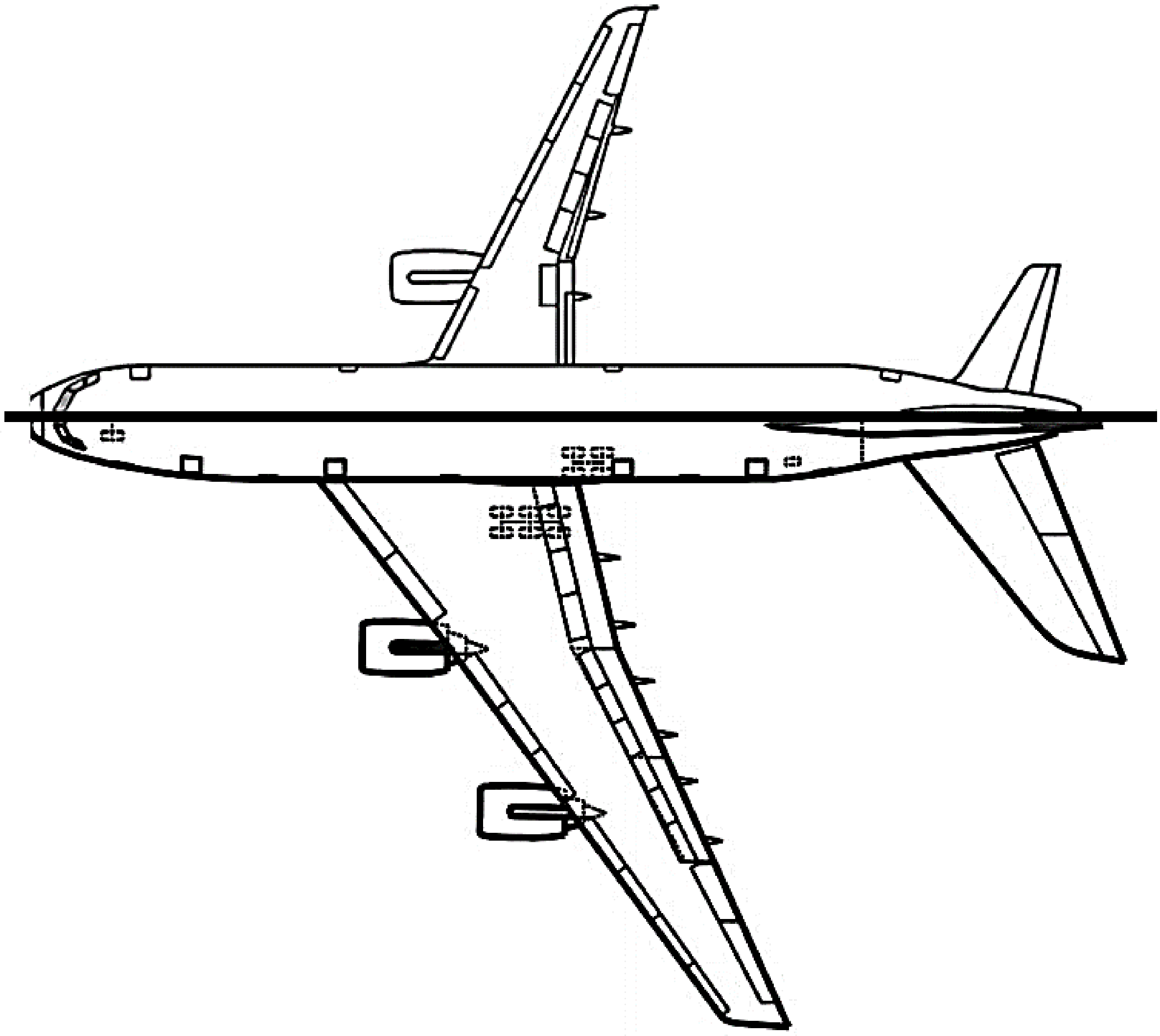

Figure 6.8 A321 scaled to the same size as the A380 (**Scholz 2006**)

But there is another problem, each aeroplane belongs to a certain category. For example, the Boeing 747-400 has a span of 64,44 m. This means that this aeroplane belongs in aircraft category E for the ICAO Aerodrome Reference Code (ARC). The upper limit of this group is 65 m. The designers of the Boeing 747 took that in consideration not to exceed this value. Because, the higher the category of the aeroplane, the fewer airports can handle the aeroplane. This span constraint means a wing surface limitation. The wing surface can not be increased by enlarging the span of the aeroplane. In the end, the wing surface is designed as large as possible, but is still small in proportion with the mass of the aircraft. This small wing surface causes a decrease in lift generation. It is important that the clean wing still produces enough lift to keep the aeroplane up during cruise. The only solution left is fly very fast, this is why the Mach number is large. The lift coefficient for take-off and landing is than achieved, using complicated high-lift systems.

The maximum lift coefficient for landing has a normal value. This is the result of a rather small landing field length in combination with a large wing loading. Keep in mind that the aeroplane has a ferry range of 8000 NM but is not an ultra-long range aircraft. This makes the relative landing mass more acceptable. This ratio is very low since this aircraft is a long range aircraft and thus loses a lot of weight during the flight. This also keeps the lift coefficient low. The maximum lift coefficient for take-off is average. For an aircraft this size, a take-off field length of 2815 m is normal. But the problem is to achieve a velocity that is large enough to encounter the small wing area and generating enough lift for take-off. Therefore, the aeroplane has to be provided with enough thrust. This thrust is provided by four engines which result in a large thrust-to-weight ratio. The aerodynamic efficiency factor is 15,1 which is a little small considering the range. This means that the maximum aerodynamic efficiency is also slightly too small. The specific fuel consumption of the Boeing 747-400 amounts up to 17,4 mg/N/s. This is a little high for a high turbofan jet engine but comes close to the typical SFC for jet powered aeroplanes (16 mg/N/s).

The Boeing 747 has a small wing reference area in proportion with its weight. To obtain sufficient lift during landing and take-off, advanced and complicated high-lift sytsems are integrated in the wings. The high-lift systems on the leading edge are Krueger flaps and variable camber slats. On the trailing edge, a three-slotted flap is mounted. In the reverse engineering program, there is no data available for three-slotted flaps and variable camber slats. An estimation is made using double-slotted flaps and nose flaps instead. The values for the maximum lift coefficient for take-off and landing are respectively 1,98 and 2,27 which is 4% smaller than the reverse engineering results. Using correct data will increase the verification values. The Boeing 747-400 has winglet features which improve the aerodynamic efficiency. The verification value for the maximum aerodynamic efficiency is 18,0 which is 7% larger than the reverse engineering value. According to **Raymer 2012**, the maximum aerodynamic efficiency of the B747 is 17,2. This deviates only 2% with the reverse engineering value. The final verification value is the specific fuel consumption. According to **Herrmann 2010** this is 16,9 mg/N/s. This is 3% less than the specific fuel consumption according to the reverse engineering.

It can be concluded that the the reverse engineering values are reliable and that the Boeing 747 has average values for the reverse engineering results. This means that size does not matter when it comes to these specifications. Only for the wing surface there are problems considering the size of the aircraft.

6.8 Blended Wing Body VELA 2

A blended wing body (BWB) is an aircraft with no clear distinction between the fuselage and the wing. The idea is to transport a large amount of passengers in an airfoil shaped fuselage which also acts as a wing. This concept can not be confused with a flying wing. Unlike the

BWB, the flying wing has no distinction between wing and fuselage at all. VELA is the abbreviation for Very Efficient Large Aircraft. This was a framework program, funded by the European Union. The goal of the research program was to investigate BWB conepts for civil aviation. It is meant to be a long range aircraft provided with four engines. The aircraft has a twin tail which is more effective than a conventional tail configurations for high angles of attack. This is because the shielding of the tail by the body is smaller. A disadvantage of this configuration is the increase in drag. Each lift force induces drag, with two tail planes, there are two lift forces and thus more induced drag. Since there is no horizontal tail, the ailerons will also act as elevators (a.k.a. elevons).

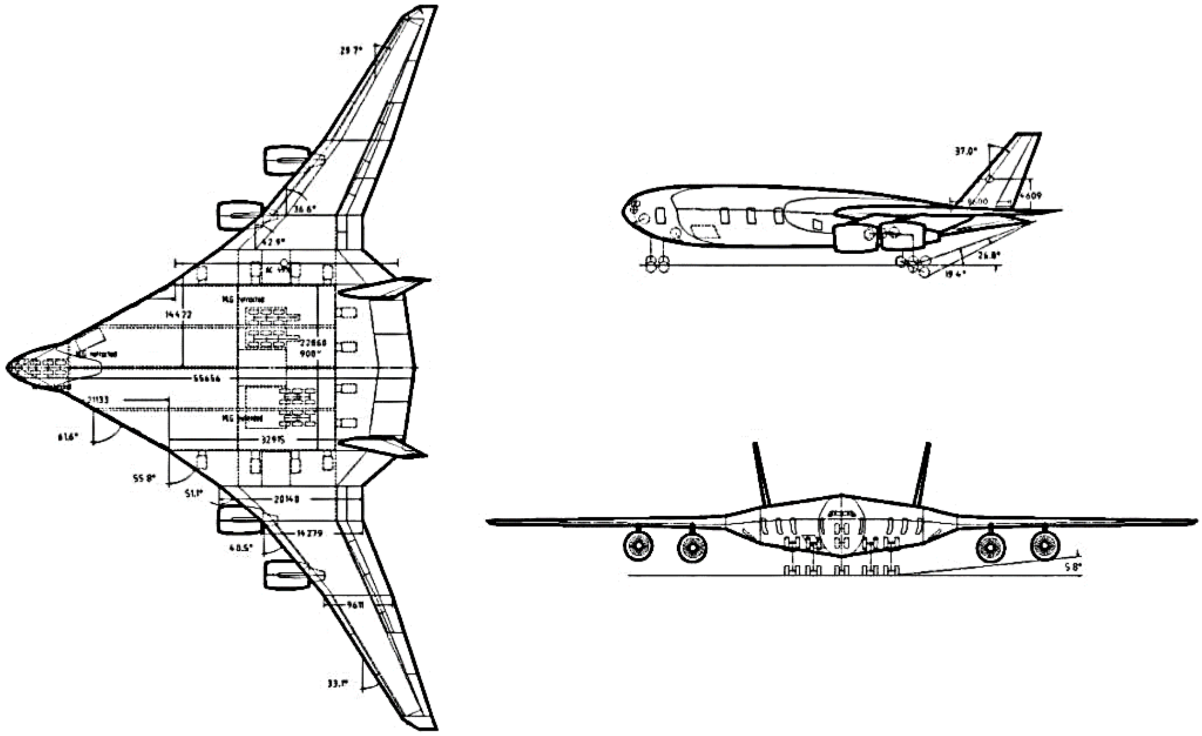

Figure 6.9 3 view drawing of the Blended Wing Body VELA 2 (**Scholz 2006**)

The 3 view drawing shows that the blended wing body requires more material to be manufactured than a standard tail-aft aeorplane. This means that the relative operating empty mass will be larger in comparison with that of a conventional aeroplane.

Table 6.12 Input values and reverse engineering results of the BWB VELA 2

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	2487	m	Large
Take-off field length	s_{TOFL}	3350	m	Large
Cruise Mach number	M_{CR}	0,85	-	Large
Design range	R	7500	NM	LR
Relative landing mass	m_{ML}/m_{MTO}	53	%	Very small
Relative operating empty mass	m_{OE}/m_{MTO}	55,1	%	Large
Wing loading	m_{MTO}/S_W	359	kg/m²	Very small
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,203	-	Very mall
Cruising altitude	h_{CR}	35000	ft	Normal
Speed ratio	V/V_{md}	1	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	0,72		Very small
Maximum lift coefficient, take-off	$C_{L,max,TO}$	1,30		Small
Maximum aerodynamic efficiency	E_{max}	25,4		Large
Specific fuel consumption	SFC	13,8	mg/N/s	Small

Because this is not a conventional aeroplane, lots of specifications are very different from average. In order to understand the large differences, it is important to know the philosophy of the blended wing body. Paragraph 6.7 explains why the wing surface of the Boeing 747 is small in comparison with its weight. This is due to the square cube law: if an object doubles in size, the surface of that object will quadruple and the volume of that object will octuple. This means that the larger the aeroplanes gets, the larger their wings get in order to be able to fly (Figure 6.8). This size growing is not unlimited because the structure still has to hold its own weight and the airports must be able to handle this aeroplane. Therefore, the blended wing body could be a solution. Its entire surface serves as a wing. This means a large wing reference area and thus a very small wing loading. This large wing surface also means a firm decrease in required lift coefficient. But a big problem is the integration of high-lift systems. Since this is very difficult, the landing field length and take-off field length are very large. The large wing surface and large field lengths make the maximum lift coefficients very small. The aeroplane is very large and heavy. It has a maximum take-off mass of 691 tons. Today's most heavy passenger aeroplane (A380, Airbus) weighs 575 tons and uses four engines resulting in a thrust-to-weight ratio of 0,221. Considering this, the heavy 691 tons BWB supplied with four engines, will have a very small thrust-to-weight ratio. Since the wing is blended into the fuselage, the drag will reduce significantly in comparison with a conventional aircraft. This reduction in drag causes an unusual large aerodynamic efficiency. Because the maximum aerodynamic efficiency is that large, the specific fuel consumption will be reduced while cruising on an average altitude at high Mach number.

Unfortunatly, it is only possible to execute a verification calculation for the specific fuel consumption. The required information and methods to calculate the maximum lift coefficient is not available and the formulas, calculating the maximum aerodynamic efficiency, do not take the smooth transition of the wing and body into account. The specific fuel consumption according to **Herrmann 2010** is 15,4 mg/N/s which deviates 12%. The overall pressure ratio and turbine entry temperature is unknown. Usually the actual OAPR is larger which causes a decrease of the fuel consumption. Therefore it is presumable that the actual error will be smaller.

6.9 Dassault Falcon 8X

The Dassault Falcon 8X is a three-engine, long range, business jet. It is a derivation of the Falcon 7X but with an increased range, better aerodynamics and a bigger fuel capacity. It is a modern aeroplane which had its first flight in 2015. The third engine is embedded in the tail. All the engines are in the back of the aeroplane. This causes a movement of the center of gravity to the back. This backshifting forces the wing to shift with the center of gravity. This shrinks the lever arm of the tail planes which results eventually in a general enlargement of the tail. Rear mounted engines are more beneficial for cabin noise but have the disadvantage to increase the operating empty weight. Looking at the aircrafts data, it stands out that the take-off field length and in particular the landing field length are very short. This means that a large maximum lift coefficient can be expected. Since the aeroplane is a new design, it can be expected that the specific fuel consumption is normal.

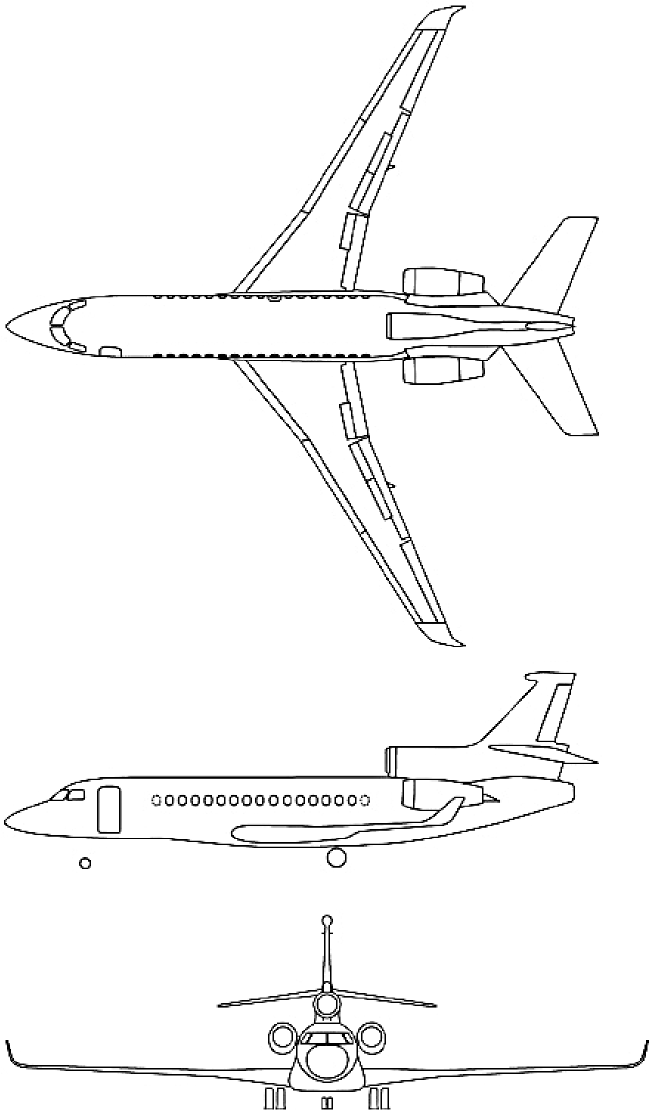

Figure 6.10 3 view drawing of the Dassault Falcon 8X (**Dassault 2017**)

Table 6.13 Input values and reverse engineering results of the Dassault Falcon 8X

Quantity name	Symbol	Value	SI Unit	Rate
Landing field length	s_{LFL}	656	m	Very small
Take-off field length	s_{TOFL}	1829	m	Normal
Cruise Mach number	M_{CR}	0,80	-	Normal
Design range	R	6450	NM	LR
Relative landing mass	m_{ML}/m_{MTO}	88,5	%	Large
Relative operating empty mass	m_{OE}/m_{MTO}	49,8	%	Normal
Wing loading	m_{MTO}/S_W	468	kg/m²	Small
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}\ g)$	0,276	-	Large
Cruising altitude	h_{CR}	37700	ft	Large
Speed ratio	V/V_{md}	1,21	-	-
Maximum lift coefficient, landing	$C_{L,max,L}$	3,70		Very large
Maximum lift coefficient, take-off	$C_{L,max,TO}$	2,17		Normal
Maximum aerodynamic efficiency	E_{max}	18,4		Small
Specific fuel consumption	SFC	18,0	mg/N/s	Large

The result for the maximum lift coefficient for landing is remarkably large. This is the result of combining a small wing loading with a large relative landing mass and a very short landing field length. The Falcon 8X is supposed to serve as a business jet. The idea is that the privat jet could land on most of the runways that are available in the world. This results in a remarkable large value for the maximum lift coefficient for landing. This also explains why the wing loading is that small. The surface of the wing is far below average because the designers increased the wing in order to achieve a larger lift during approach. The surface of the wing is large in proportion with its maximum take-off mass. This is decisive in order to meet the requirements of a short landing field length. If an average wing loading was obtained, the maximum lift coefficient for landing would be 4,5 which is impossible to achieve using high-lift systems. The reverse engineering value for the maximum lift coefficient for take-off is on the edge of normal. This is the result of combining a normal take-off field length with a large thrust-to-weight ratio. If the Falcon 8X only had two engines, the lift coefficient or the take-off field length would have to increase. Therefore, the three engine configuration is essential. The aerodynamic efficiency factor is 14,2 and differs a lot from the 17,25 reference value. This means that also the maximum aerodynamic efficiency is rather small. The Falcon 8X flies high and fast. This high altitude decreases the specific fuel consumption, but the high cruising speed increases it to a rather large value of 18,0 mg/N/s.

The Falcon 8X has a double-slotted flap over 50% of the structural span and for 85% Handly Page slats mounted. Unfortunately, more accurate information is not available. The verification value for the maximum lift coefficient is therefore unreliable. The values are 2,17 and 1,17 for respectively landing and take-off. This deviates 41% from the reverse engineering values. Furthermore, the aeroplane has winglets with an individual height of 1,05 m. This has an effect on the maximum aerodynamic efficiency. The theoretical approach gives a value of 22,3 which is 21% larger than the maximum aerodynamic efficiency according to the reverse engineering. The winglets also have a positive effect on the specific fuel consumption. The verification value is very large (26,1 mg/N/s). It differs 45% from the reverse engineering value for the SFC. According to **Jenkinson 2017b**, the SFC is 19,1 mg/N/s. This only deviates with 6% from the reverse engineering result. All the values which describes the characteristics of the aeroplane diverge a lot from the theoretically determined values while the deviations of the specifications with respect to the actual aeroplane are almost zero.

Just like the Bae 146, it can be concluded that the aeroplane requirements are too unique so that it is impossible to determine a correct verification value with formulas which are based on empirical data. These formulas represent the average of aeroplanes. They are only a good estimation when the requirements have normal average values. This does not apply to the verification calculations for the maximum lift coefficients. Too much parameters are unknown about the wing and the high-lift systems so that it is impossible to make a good theoretical estimation. This is self-evident since the Falcon 8X is a very recent design.

6.10 Conclusion

This chapter contains graphs that include all the investigated and reverse engineered aeroplanes in chronological order. This gives an overview of the evolution of certain parameters in aircraft history. The graphs are not always smooth and not every aeroplane seems to fit in the picture. Paramount are the requirements of the aeroplane. Some aeroplanes are designed for a very specific purpose. This results in deviating parameters. Keep also in mind that the Blended Wing Body VELA 2, the Boeing SUGAR High and the Rebel are case studies. In some aspects, they are futuristic or innovative in comparison with the conventional aeroplanes and therefore, they will deviate in many ways. A chronological list of the aeroplanes is shown in the following table:

Table 6.14 Investigated and reverse engineered aeroplanes in chronological order

Year of first flight	Aircraft
1955	Caravelle 10B
1958	Boeing 707-320C
1982	BAe 146-200
1987	Boeing 747-400
1989	A320-200
2006	BWB VELA 2
2010	Boeing SUGAR High
2014	The Rebel
2015	Dassault Falcon 8X

The maximum lift coefficient is depending on the requirements of the aeroplane. The lift coefficient for landing is determined by the landing field length, the wing loading and the relative landing mass. The maximum lift coefficient for take-off is determined by the thrust-to-weight ratio and the take-off field length. Short field lengths result in high lift coefficients. This has a big influence on the value for the maximum lift coefficient and is an airport performance requirement. Therefore, it is not relevant to plot the maximum lift coefficient chronologically.

Parameters that are interesting to plot chronologically are the maximum aerodynamic efficiency, the specific fuel consumption, the wing loading, the thrust-to-weight ratio and the cruise altitude.

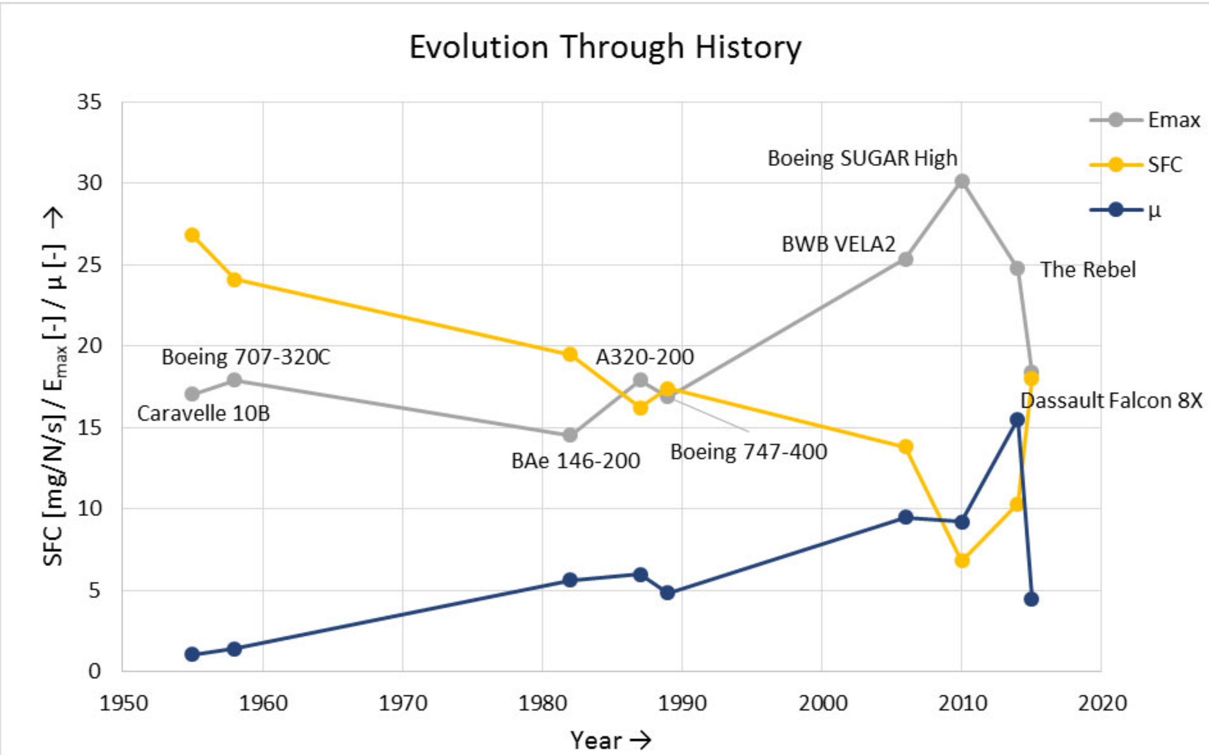

Figure 6.11 Evolution of the maximum aerodynamic efficiency and the specific fuel consumption

In the year 1980, the price of kerosene knew for the first time in history a huge increase. The price got more than doubled. This was an important factor for the airline industry to develop and use more economical engines. Figure 6.11 shows what has changed in time and what causes the decrease of fuel consumption.

The first positive influence on the fuel consumption is due to the engines. The engine technology improved and the bypass ratio enlarged. But the changing in the curves from the specific fuel consumption and bypass ratio show that the change is not always parallel. This means that another factor is involved.

The second factor is the aerodynamic efficiency of the aeroplanes. The engine manufacturers are not the only ones who can improve the fuel consumption. The design of the aeroplanes got better, the use of winglets increased the maximum aerodynamic efficiency and this results in a better fuel consumption. It is obvious that the maximum aerodynamic efficiency of aeroplanes have increased. This causes a decrease from the specific fuel consumption.

Improved engines and better aircraft design are not the only factors that influences the fuel consumption. Figure 6.12 shows that a higher cruise altitude has a positive effect on the specific fuel consumption.

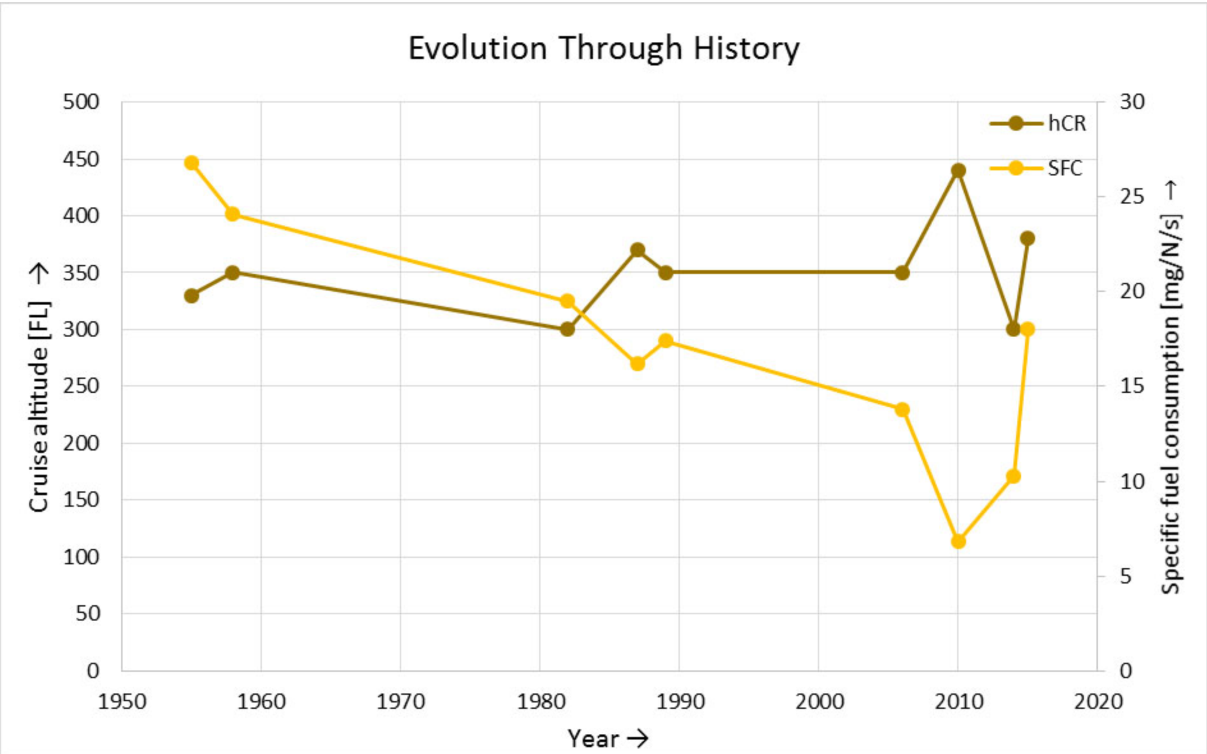

Figure 6.12 Evolution of the cruise altitude and the specific fuel consumption

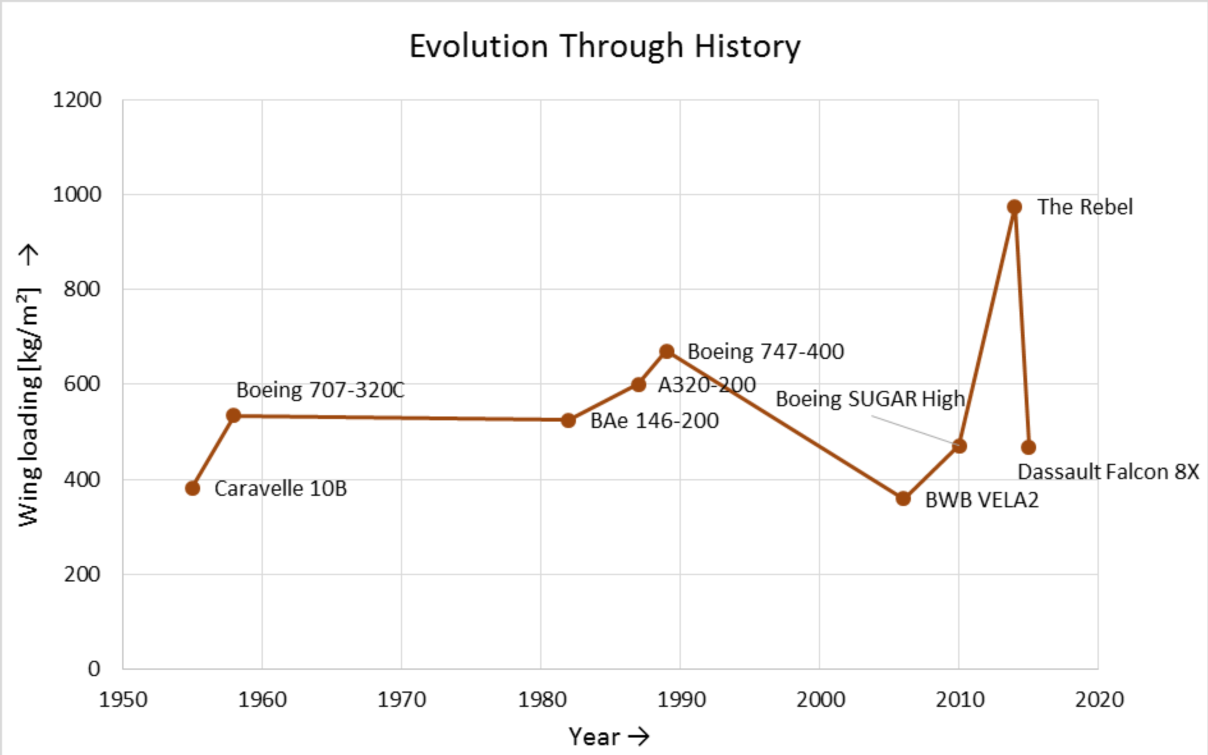

Figure 6.13 Evolution of the wing loading

A typical value for the wing loading is 586 kg/m² (according to **Raymer 1989**). The graph shows that most of the passenger jets fluctuate around this value. The Caravelle and the BWB have a very low value for the wing loading. The A320, the Boeing 747 and the Rebel have a very large wing loading. It can be said that the wing loading does not change in time. It depends solely on the design and requirements. **Raymer 1989** is correct in his assumption that the wing loading has an average for jet transport.

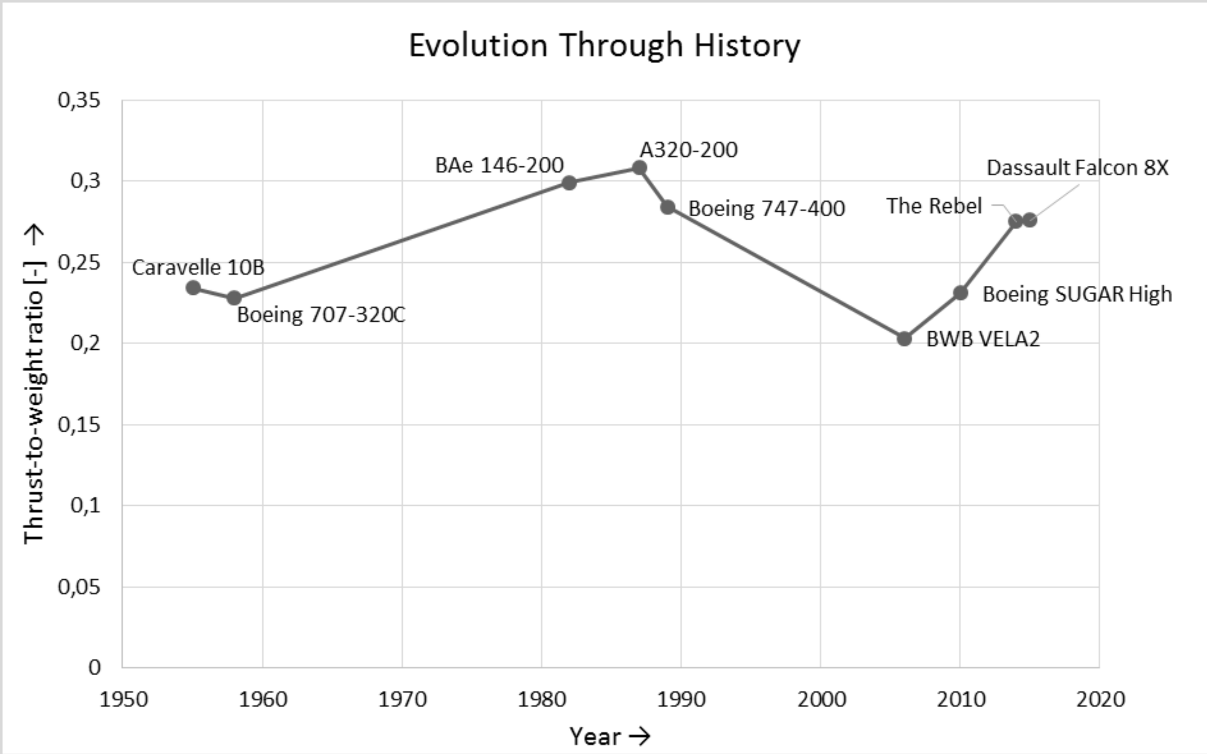

Figure 6.14 Evolution of the thrust-to-weight ratio

For the thrust-to-weight ratio, **Raymer 1989** also has a typical average of 0,25. This has not changed significantly through history. The graph shows that jet transport exceeds the 0,25 to a certain degree only since 1980 (not including the study cases of the BWB, the Boeing SUGAR High and the Rebel). A possible reason for this is the fact that aeroplanes are getting larger and heavier. This means that they need more take-off field length. But if the aeroplanes have to be accepted by as many airports as possible, then they have to minimize the take-off field length. This can be done by enlarging the maximum lift coefficient for take-off or by increasing the thrust to weight ratio. This conclusion is uncertain because the amount of investigated aeroplanes is too small.

It can be concluded that the reverse engineering tool is an excellent method to investigate the evolution of aeroplanes. Better results can be obtained by performing the reverse engineering for a larger number of passenger jets. Furthermore, this tool can also be used to compare different aeroplanes or to use the classified design parameters in other works, investigations or calculations.

7 Recommendations

The speed ratio is essential when it comes to determining the maximum aerodynamic efficiency and specific fuel consumption. Therefore, it is important that the correct speed ratio is applied in the calculations. At this stage, the program will change this ratio between 1 and 1,316. The value is determined according to the error that is made with the original cruise altitude and cruise speed. In most cases, the speed ratio will be either 1 or 1,316. If the program determines a speed ratio that is close to these values, the error with the original values will be minimized, but it is possible that the actual speed ratio is 1 or 1,316. It is up to the user to oblige the program to apply this value for the reverse engineering. This is done by changing the upper limit to 1 or the lower limit to 1,316 (depending on the required value for the speed ratio). In this thesis, it is applied to the reverse engineering of the Boeing 707-320C, the A320-200, the Rebel, the Boeing SUGAR High and the Boeing 747-400.

For most conventional aircrafts, the design point is determined by the landing and take-off constraints. The other parts of the flight mission (second segment, cruise and missed approach) are fit as close to the design point as possible in order to obtain the perfect aircraft design. But this means that, for example, the cruise curve matches with the design point. If the user knows the equivalent thrust-to-weight ratio during cruise, it can be changed in the tab '2) E_max'. This results in a cruise curve which is separated from the design point. This will give a more accurate result to the user. Another possibility is to do research for additional information. Some sources provide values for the take-off thrust and the cruise thrust (for example **Jenkinson 2017b**). Knowing this information and equation (2.15), a solution might be possible to separate the cruise curve from the design point.

There are always possibilities for improvement. Some potential recommendations to improve this program and exclude possible shortcommings are:

- Integrate a function that does the reverse engineering for either the minimum drag speed or the maximum range speed, depending on the value for the optimized speed ratio by the program.
- Integrate a possibility to separate the cruise curve from the design point.

8 Summary

The goal of this thesis is to determine the classified design parameters of passenger jets. These classified design parameters are the maximum lift coefficient for landing and take-off, the maximum aerodynamic efficiency and the specific fuel consumption. The formulas to calculate these parameters are all based on the preliminary sizing of jet aircrafts. By analysing this, it is possible to know the mutual relations of the aircraft specifications and the design parameters.

The philosophy of reverse engineering is to investigate a design and return to the fundamentals of the product. This is done by reversing the steps of the original designer. By doing so, the design parameters of a product can be determined. Applying this to the preliminary sizing, formulas are obtained to determine the design parameters in function of the aircraft specifications. For the maximum lift coefficient for landing, this is done by applying equation (4.1). For take-off this is equation (4.2). The maximum aerodynamic efficiency is calculated using formula (4.9). This equation is complex and is solved through numerical iterations. The last design parameter, the specific fuel consumption, is determined with equation (4.11) or (4.18).

Since these formulas and mutual relations are circuitous, a program is made so that a user, with basic knowledge of aircraft technology, is able to perform the reverse engineering. The tool is made user friendly by minimizing the number of parameters and shows the user notifications on tricky situations. The tool is relatively fast; most of the time goes to aircraft specification research. The tool is also multifunctional because the program allows to have certain parameters unknown. The user can choose which range is used and in function of this choice, the program will apply different formulas. The program also plots the matching chart of the aeroplane. This tool meets the requirements of user friendly, quick, multifunctional and the user has the possibility for a direct print-out of the results.

Because the design parameters are classified, it is difficult to judge the accuracy and reliability of the program. Most cerainly it all starts with the accuracy and reliability of the inputs and thus the source. The program itself is not flawless because of the following facts:

- All calculations are based on preliminary sizing, which is a rudimentary method in the designing process. This disadvantageous the accuracy. It becomes a matter of choosing between accuracy and usability.
- Possible errors because the design point is used to calculate the design parameters. The more the actual cruise curve deviates from the design point, the more the reverse engineering results will deviate from the original design.
- At this stage, the program will change the speed ratio between 1 and 1,316. If a wrong speed ratio was determined by the program, it is up to the user to notice and correct this.
- Deviations for the specific fuel consumption because the actual mission fuel fractions deviate from the average mission fuel fractions for jet transport or business jets.

In general, it can be said that the tool is accurate and reliable. But the user has to be critical and pay attention because it is very easy to make mistakes.

The program appears to be an excellent working tool which makes it possible to understand the philosophy behind the design of an aircraft and to understand what has changed during aircraft history. It gives an image of the progress human kind made, starting from Otto Lilienthal and the Wright brothers untill today's aeroplanes. Even the more futuristic concepts can be handled by the program with a good accuracy.

The result of this thesis is a program that is able to determine the maximum lift coefficient for landing and take-off, the maximum aerodynamic efficiency and the specific fuel consumption of a designed aeroplane. This tool can be used by a person who knows basic aircraft technology. It is also multifunctional and fast. The results are reliable and accurate for any scientist that is self critical.

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Appendix A Caravelle 10B (Sud-Aviation)

Aeroplane Specifications

Data to apply reverse engineering

					LL	*UL*
Landing field length	**Known**	s_{LFL}	**1707**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	70,3	70,3
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2134**	m	2134	2134
Temperature above ISA (288,15K)		ΔT_{TO}	**15**	K		
Relative density		σ	**0,951**			
Range (maximum PAX)		R	**1431**	NM		
Cruise Mach number		M_{CR}	**0,765**			
Wing area		S_W	**147**	m²		
Wing span	**Known**	b_W	**34,3**	m²	34,3	34,3
Aspect ratio		A	8,02			
Maximum take-off mass		m_{MTO}	**56000**	kg		
Payload mass (maximum PAX)		m_{PL}	**9100**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,163			
Maximum landing mass		m_{ML}	**53200**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,950			
Operating empty mass		m_{OE}	**30055**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,537			
Wing loading		m_{MTO}/S_W	381,7	kg/m²		
Number of engines		n_E	**2**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**64,4**	kN		
Total take-off thrust		T_{TO}	128,8	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,234			
Bypass ratio		μ	**1,06**			

Data to optimize V/V_{md}

					LL	UL
Cruise speed	**Known**	V_{CR}	**228**	m/s	228	228
Cruise altitude	**Known**	h_{CR}	**10000**	m	10000	10000
Speed ratio		V/V_{md}	**1,316**	-	1	1,316

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**20**	°		
Mean aerodynamic chord		c_{MAC}	**5,3**	m		
Position of maximum camber		$x_{(y\_c),max}$	**15**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**2**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**12,0**	%		
Taper		λ	**0,354**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1707	1707	m	0,00%
Approach speed	V_{APP}	Unknown	**70,3**	m/s	0,00%
Take-off field length	s_{TOFL}	2134	2134	m	0,00%
Span	b_W	34,3	34,3	m	0,00%
Aspect ratio	A	8,02	8,02		0,00%
Cruise speed	V_{CR}	227,6	228	m/s	0,12%
Cruise altitude	h_{CR}	10000,0	10370	m	3,70%
Squared Sum					**1,37E-03**
Absolute maximum deviation					3,7%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**1,99**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,88**	
Maximum aerodynamic efficiency	E_{max}	**17,05**	
Specific fuel consumption	SFC	**2,68E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing			
Landing field length	s_{LFL}	1707	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	70,31	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,95	
Wing loading at maximum take-off mass	m_{MTO}/S_W	381,7	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**1,99**	
Take-off			
Take-off field length	s_{TOFL}	2134	m
Temperatur above ISA (288,15K)	ΔT_{TO}	15	K
Relative density	σ	0,95	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,234	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,88**	
2nd Segment			
Aspect ratio	A	8,020	
Lift coefficient, take-off	$C_{L,TO}$	1,30	
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020	
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,010	
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000	
Profile drag coefficient	$C_{D,P}$	0,030	
Oswald efficiency factor; landing configuration	e	0,7	
Aerodynamic efficiency in take-off configuration	E_{TO}	10,30	
Number of engines	n_E	2	
Climb gradient	$\sin(\gamma)$	0,024	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,242	
Missed approach			
Lift coefficient, landing	$C_{L,L}$	1,17	
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020	
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,004	
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000	
Choose: Certification basis	JAR-25 resp. CS-25	**no**	
	FAR Part 25	**yes**	
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015	
Profile drag coefficient	$C_{D,P}$	0,039	
Aerodynamic efficiency in landing configuration	E_L	10,04	
Climb gradient	$\sin(\gamma)$	0,021	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,229	

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,765	
Aspect ratio	A	8,02	
Bypass ratio	μ	1,06	
Wing loading	m_{MTO}/S_W	382	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,234	

Variables

	V/V_{md}	**1,3**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,018
Lift coefficient at E_{max}	$C_{L,md}$	0,63
Ratio, lift coefficient	$C_L/C_{L,md}$	0,577
Lift coefficient, cruise	C_L	0,363
Actual aerodynamic efficiency, cruise	E	14,77
Max. aerodynamic efficiency, cruise	E_{max}	**17,05**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,11	0,00	0,00
f'(x)	-0,11	-0,11	-0,11
E_{max}	16	17,07	17,05

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	1431	NM
Mach number, cruise	M_{CR}	0,765	
Bypass ratio	μ	1,06	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,234	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	56000	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,163	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,537	

Calculated values			
Actual aerodynamic efficiency, cruise	E	14,77	
Cruise altitude	h_{CR}	10370	m
Cruise speed	V_{CR}	228	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,301	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,699	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	370400	m
Loiter time	t_{loiter}	2700	s
Specific fuel consumption	SFC	**2,68E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 65 series**	
Wing span	b_W	34,3	m
Structural wing span	$b_{W,struct}$	36,50	m
Wing area	S_W	146,7	m²
Aspect ratio	A	8,02	
Sweep	φ_{25}	20	°
Mean aerodynamic chord	c_{MAC}	5,3	m
Position of maximum camber	$x_{(y\_c),max}$	15	%c
Camber	$(y_c)_{max}/c$	2	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	12,0	%
Taper	λ	0,354	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	70,31	m/s
Mach number, landing	M_{APP}	0,21	
Mach number, cruise	M_{CR}	0,765	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,3	%c
Leading edge sweep	φ_{LE}	23,4	°
Reynoldsnumber	Re	2,7E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,35	
Correction term, camber	$\Delta_1 c_{L,max}$	0,25	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,021	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,623	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,85	
Correction term, Mach number	$\Delta C_{L,max}$	-0,01	
Lift coefficient, wing	$C_{L,max}$	1,38	
Calculations increase of lift coefficient due to flaps		**1 flap type**	
Correction factor, sweep	K_φ	0,89	
• Flap group A			
Single-slotted flap	$\Delta c_{L,max,fA}$	0,74	
Use flapped span	b_W,fA	**24,3**	m
Percentage of flaps allong the wing		67%	

Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,43
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,43

Calculations increase of lift coefficient due to slats — **1 slat type**

Sweep angle of the hinge line	$\varphi_{H.L.}$	**26**	°
• Slat group A			
Fixed slat	$\Delta c_{L,max,sA}$	0,51	
Use slatted span	b_W,sA	**17,63**	m
Percentage of slats allong the wing		48%	
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,22	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,22	

Wing

Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,01**
RE value maximum lift coefficient, landing		1,99
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,90**
RE value maximum lift coefficient, take-off		1,88
1%		

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
No winglets	$k_{e,WL}$	1,00
Aspect ratio	A	8,02
Effective aspect ratio	A_{eff}	8,02
Efficiency factor, short range	k_E	15,15
Relative wetted area	S_{wet}/S_W	**6,10**
Verification value maximum aerodynamic efficiency	E_{max}	**17,4**
RE value maximum aerodynamic efficiency		17,05
2%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,765	
Cruise altitude	h_{CR}	10000	m
Bypass Ratio	μ	1,06	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	64,40	kN
Overall Pressure ratio	OAPR	**15,40**	
Turbine entry temperature	TET	**1395,78**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,97	
Ventilator efficiency	$\eta_{ventilator}$	0,74	
Compressor efficiency	$\eta_{compresor}$	0,77	
Turbine efficiency	$\eta_{turbine}$	0,88	
Nozzle efficiency	η_{nozzle}	0,97	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	223,15	K
Dimensionless turbine entry temperature	ϕ	6,25	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,12	
Temperature function	χ	1,32	
Gas generator efficiency	η_{gasgen}	0,99	
Gas generator function	G	1,80	
Verification value specific fuel consumption	SFC	0,87	kg/daN/h
Verification value specific fuel consumption	SFC	**2,41E-05**	kg/N/s
RE value specific fuel consumption	SFC	2,68E-05	kg/N/s

-10%

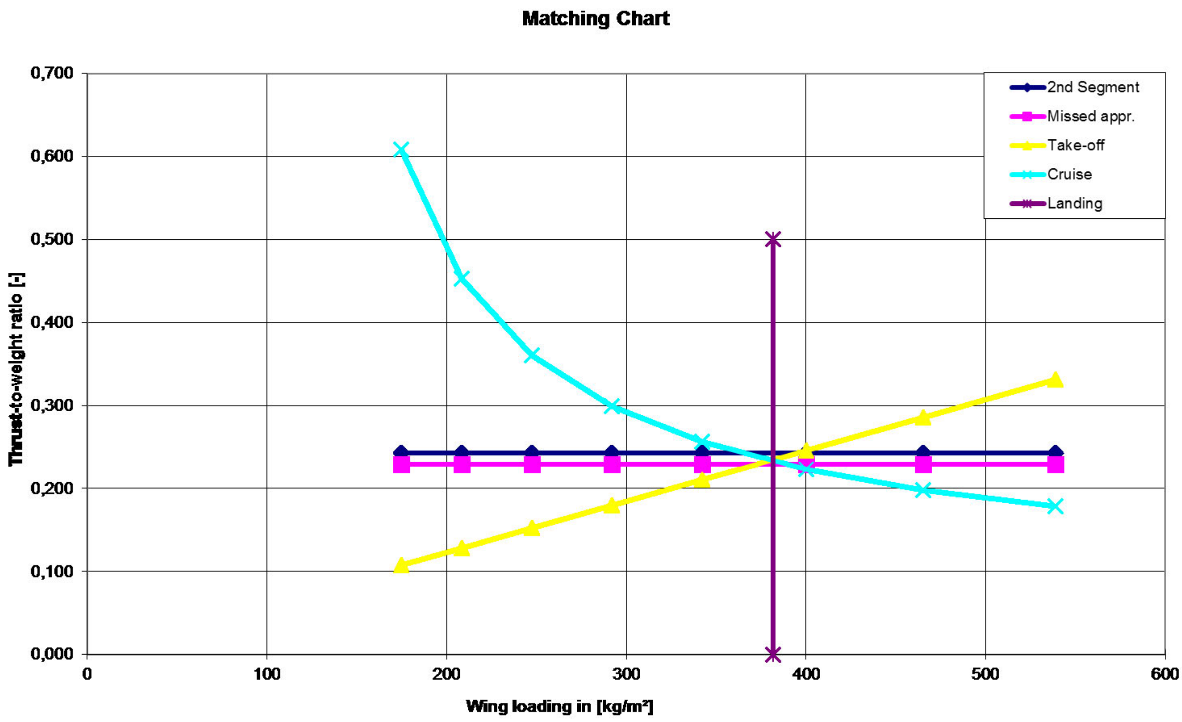

Figure A.1 Matching chart – Caravelle 10B

Appendix B Boeing 707-320C

B.1 Optimized Speed Ratio

Aeroplane Specifications

Data to apply reverse engineering

					LL	*UL*
Landing field length	**Known**	s_{LFL}	**1905**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	74,3	74,3
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**3054**	m	3054	3054
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Maximum range		R	**5000**	NM		
Cruise Mach number		M_{CR}	**0,82**			
Wing area		S_W	**283**	m²		
Wing span	**Known**	b_W	**44,42**	m²	44,42	44,42
Aspect ratio		A	6,96			
Maximum take-off mass		m_{MTO}	**151315**	kg		
Payload mass (maximum range)		m_{PL}	**12852**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,085			
Maximum landing mass		m_{ML}	**112037**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,740			
Operating empty mass		m_{OE}	**66224**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,438			
Wing loading		m_{MTO}/S_W	533,9	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**84,5**	kN		
Total take-off thrust		T_{TO}	338	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,228			
Bypass ratio		μ	**1,43**			
Available fuel volume		$V_{fuel,available}$	**90,299**	m³		

Data to optimize V/V_{md}

					LL	*UL*
Cruise speed		V_{CR}	**246**	m/s		
Cruise altitude		h_{CR}	**10668**	m		
Speed ratio		V/V_{md}	**1,110**	-	1	1,316

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**35**	°		
Mean aerodynamic chord		c_{MAC}	**7,36**	m		
Position of maximum camber		$x_{(y\_c),max}$	**15**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**4**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**10,0**	%		
Taper		λ	**0,259**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	*RE value*	*Unit*	*Deviation*
Landing field length	s_{LFL}	1905	1905	m	0,00%
Approach speed	V_{APP}	Unknown	**74,3**	m/s	0,00%
Take-off field length	s_{TOFL}	3054	3054	m	0,00%
Span	b_W	44,42	44,42	m	0,00%
Aspect ratio	A	6,96	6,96		0,00%
Cruise speed	V_{CR}	245,9	243	m/s	-1,07%
Cruise altitude	h_{CR}	10668,0	10650	m	-0,17%
Squared Sum					**1,16E-04**
Absolute maximum deviation					1,1%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**1,94**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,80**	
Maximum aerodynamic efficiency	E_{max}	**16,38**	
Specific fuel consumption (acc. to PL and OE)	SFC	**2,17E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**2,10E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1905	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	74,28	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,74	
Wing loading at maximum take-off mass	m_{MTO}/S_W	533,9	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**1,94**	

Take-off

Take-off field length	s_{TOFL}	3054	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,80**	

2nd Segment

Aspect ratio	A	6,962
Lift coefficient, take-off	$C_{L,TO}$	1,25
Lift-independent drag coefficient, clean	$C_{D,0}$ (2^{nd} Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,007
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,027
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	9,67
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,178

Missed approach

Lift coefficient, landing	$C_{L,L}$	1,15
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,002
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,037
Aerodynamic efficiency in landing configuration	E_L	9,30
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,133

2) Maximum Aerodynamic Efficiency

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications			
Mach number, cruise	M_{CR}	0,82	
Aspect ratio	A	6,96	
Bypass ratio	μ	1,43	
Wing loading	m_{MTO}/S_W	534	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	

Variables		
	V/V_{md}	**1,1**

Calculations		
Zero-lift drag coefficient	$C_{D,0}$	0,017
Lift coefficient at E_{max}	$C_{L,md}$	0,57
Ratio, lift coefficient	$C_L/C_{L,md}$	0,812
Lift coefficient, cruise	C_L	0,461
Actual aerodynamic efficiency, cruise	E	16,03
Max. aerodynamic efficiency, cruise	E_{max}	**16,38**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,04	0,00	0,00
f'(x)	-0,11	-0,12	-0,12
E_{max}	16	16,38	16,38

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	5000	NM
Mach number, cruise	M_{CR}	0,82	
Bypass ratio	μ	1,43	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	
Available fuel volume	$V_{fuel,available}$	90,299	m³
Maximum take-off mass	m_{MTO}	151315	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,085	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,438	

Calculated values			
Actual aerodynamic efficiency, cruise	E	16,03	
Cruise altitude	h_{CR}	10650	m
Cruise speed	V_{CR}	243	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations

Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,477	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,523	
Available fuel mass	$m_{F,available}$	72239,2	kg
Relative fuel mass (acc. to fuel capacity)	$m_{F,available}/m_{MTO}$	0,477	
Mission fuel fraction (acc. to fuel capacity)	M_{ff}	0,533	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**no**	
	international	**yes**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	833400	m
Loiter time	t_{loiter}	1800	s
Specific fuel consumption (acc. to PL and OE)	SFC	**2,17E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**2,10E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	44,42	m
Structural wing span	$b_{W,struct}$	54,23	m
Wing area	S_W	283,4	m²
Aspect ratio	A	6,96	
Sweep	φ_{25}	35	°
Mean aerodynamic chord	c_{MAC}	7,36	m
Position of maximum camber	$x_{(y\_c),max}$	15	%c
Camber	$(y_c)_{max}/c$	4	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	10,0	%
Taper	λ	0,259	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	74,28	m/s
Mach number, landing	M_{APP}	0,22	
Mach number, cruise	M_{CR}	0,82	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,6	%c
Leading edge sweep	φ_{LE}	39,8	°
Reynoldsnumber	Re	3,9E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,47	
Correction term, camber	$\Delta_1 c_{L,max}$	0,27	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,022	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,761	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,73	
Correction term, Mach number	$\Delta C_{L,max}$	-0,01	
Lift coefficient, wing	$C_{L,max}$	1,28	

Calculations increase of lift coefficient due to flaps			**2 flap types**
Correction factor, sweep	K_{φ}	0,81	
• Flap group A			
Double-slotted flap	$\Delta c_{L,max,fA}$	1,25	
Use flapped span	b_W,fA	**19**	m
Percentage of flaps allong the wing		35%	
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,36	
• Flap group B			
0,3c Single-slotted fowler flap	$\Delta c_{L,max,fB}$	1,52	
Use flapped span	b_W,fB	**3,8**	m
Percentage of flaps allong the wing		7%	
Increase in maximum lift coefficient, flap group B	$\Delta C_{L,max,fB}$	0,09	
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,44	

Calculations increase of lift coefficient due to slats			**1 slat type**
Sweep angle of the hinge line	$\varphi_{H.L.}$	**38**	°
• Slat group A			
Handley Page slat	$\Delta c_{L,max,sA}$	0,55	
Use slatted span	b_W,sA	**41,8**	m
Percentage of slats allong the wing		77%	
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,33	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,33	

Wing

Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,03**
RE value maximum lift coefficient, landing		1,94
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,89**
RE value maximum lift coefficient, take-off		1,80
5%		

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
No winglets	$k_{e,WL}$	1,00
Aspect ratio	A	6,96
Effective aspect ratio	A_{eff}	6,96
Efficiency factor, short range	k_E	16,19
Relative wetted area	S_{wet}/S_W	**6,20**
Verification value maximum aerodynamic efficiency	E_{max}	**17,2**
RE value maximum aerodynamic efficiency		16,38
5%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,820	
Cruise altitude	h_{CR}	10668	m
By Pass Ratio	μ	1,43	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	84,50	kN
Overall Pressure ratio	OAPR	**13,50**	
Turbine entry temperature	TET	**1425,33**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,97	
Ventilator efficiency	$\eta_{ventilator}$	0,78	
Compressor efficiency	$\eta_{compresor}$	0,80	
Turbine efficiency	$\eta_{turbine}$	0,88	
Nozzle efficiency	η_{nozzle}	0,98	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	218,81	K
Dimensionless turbine entry temperature	ϕ	6,51	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,13	
Temperature function	χ	1,25	
Gas generator efficiency	η_{gasgen}	0,99	
Gas generator function	G	2,05	
Verification value specific fuel consumption	SFC	0,86	kg/daN/h
Verification value specific fuel consumption	SFC	**2,38E-05**	kg/N/s
RE value specific fuel consumption (acc. to PL and OE)	SFC	2,17E-05	kg/N/s
9%			
RE value specific fuel consumption (acc. to fuel capacity)	SFC	2,1E-05	kg/N/s
13%			

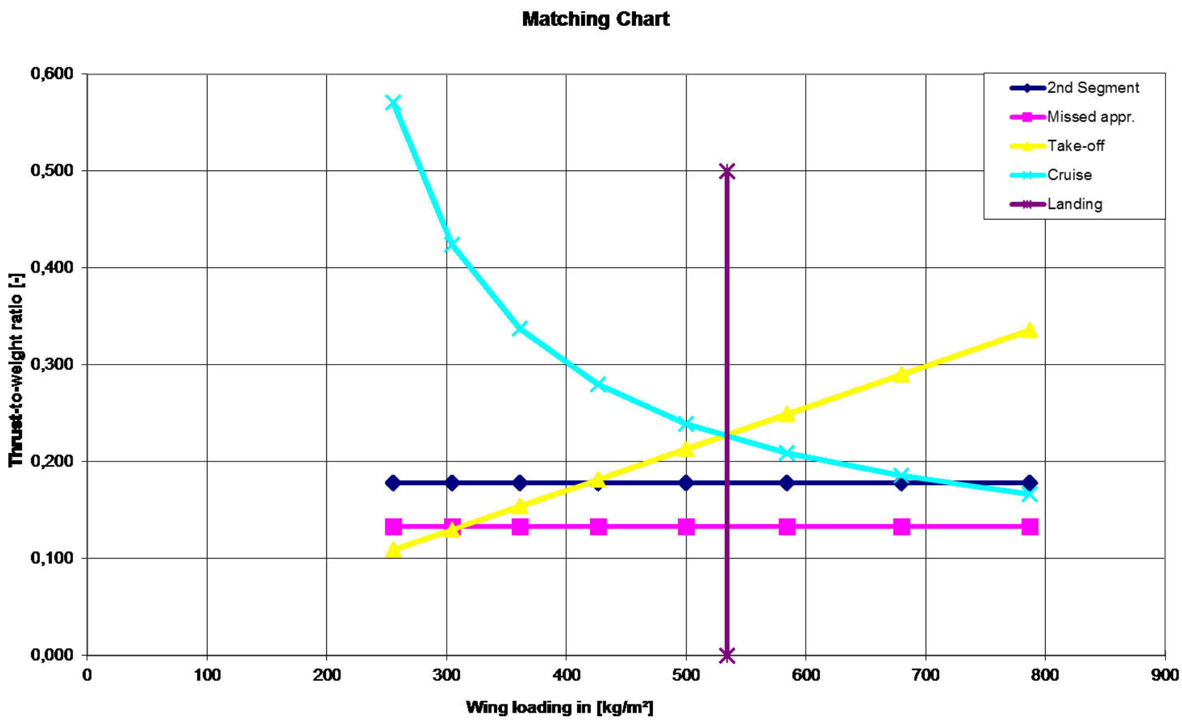

Figure B.1 Matching chart – Boeing 707-320C (optimized speed ratio)

B.2 Minimum Drag Speed

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**1905**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	74,3	74,3
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**3054**	m	3054	3054
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Maximum range		R	**5000**	NM		
Cruise Mach number		M_{CR}	**0,82**			
Wing area		S_W	**283**	m²		
Wing span	**Known**	b_W	**44,42**	m²	44,42	44,42
Aspect ratio		A	6,96			
Maximum take-off mass		m_{MTO}	**151315**	kg		
Payload mass (maximum range)		m_{PL}	**12852**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,085			
Maximum landing mass		m_{ML}	**112037**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,740			
Operating empty mass		m_{OE}	**66224**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,438			
Wing loading		m_{MTO}/S_W	533,9	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**84,5**	kN		
Total take-off thrust		T_{TO}	338	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,228			
Bypass ratio		μ	**1,43**			
Available fuel volume		$V_{fuel,available}$	**90,299**	m³		

Data to optimize V/V_{md}

				LL	UL
Cruise speed	V_{CR}	**246**	m/s		
Cruise altitude	h_{CR}	**10668**	m		
Speed ratio	V/V_{md}	**1,000**	-	1	**1,0**

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**35**	°		
Mean aerodynamic chord		c_{MAC}	**7,36**	m		
Position of maximum camber		$x_{(y\_c),max}$	**15**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**4**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**10,0**	%		
Taper		λ	**0,259**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1905	1905	m	0,00%
Approach speed	V_{APP}	Unknown	**74,3**	m/s	0,00%
Take-off field length	s_{TOFL}	3054	3054	m	0,00%
Span	b_W	44,42	44,42	m	0,00%
Aspect ratio	A	6,96	6,96		0,00%
Cruise speed	V_{CR}	245,9	242	m/s	-1,58%
Cruise altitude	h_{CR}	10668,0	11408	m	6,94%
Squared Sum					**5,06E-03**
Absolute maximum deviation					6,9%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**1,94**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,80**	
Maximum aerodynamic efficiency	E_{max}	**17,90**	
Specific fuel consumption (acc. to PL and OE)	SFC	**2,41E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**2,33E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1905	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	74,28	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,74	
Wing loading at maximum take-off mass	m_{MTO}/S_W	533,9	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**1,94**	

Take-off

Take-off field length	s_{TOFL}	3054	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,80**	

2nd Segment

Aspect ratio	A	6,962
Lift coefficient, take-off	$C_{L,TO}$	1,25
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,007
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,027
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	9,67
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,178

Missed approach

Lift coefficient, landing	$C_{L,L}$	1,15
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,002
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,037
Aerodynamic efficiency in landing configuration	E_L	9,30
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,133

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,82	
Aspect ratio	A	6,96	
Bypass ratio	μ	1,43	
Wing loading	m_{MTO}/S_W	534	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	

Variables

	V/V_{md}	**1,0**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,015
Lift coefficient at E_{max}	$C_{L,md}$	0,52
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	0,519
Actual aerodynamic efficiency, cruise	E	17,90
Max. aerodynamic efficiency, cruise	E_{max}	**17,90**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,20	-0,01	0,00
f'(x)	-0,10	-0,11	-0,11
E_{max}	16	17,97	17,90

3) Specific Fuel Consumption

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications

Range	R	5000	NM
Mach number, cruise	M_{CR}	0,82	
Bypass ratio	μ	1,43	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,228	
Available fuel volume	$V_{fuel,available}$	90,299	m³
Maximum take-off mass	m_{MTO}	151315	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,085	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,438	

Calculated values

Actual aerodynamic efficiency, cruise	E	17,90	
Cruise altitude	h_{CR}	11408	m
Cruise speed	V_{CR}	242	m/s

Mission fuel fraction

Type of aeroplane (according to Roskam)	**Transport jet**	
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980
Fuel-Fraction, descent	$M_{ff,DES}$	0,990
Fuel-Fraction, landing	$M_{ff,L}$	0,992

Calculations

Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,477	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,523	
Available fuel mass	$m_{F,available}$	72239,2	kg
Relative fuel mass (acc. to fuel capacity)	$m_{F,available}/m_{MTO}$	0,477	
Mission fuel fraction (acc. to fuel capacity)	M_{ff}	0,533	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**no**	
	international	**yes**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	833400	m
Loiter time	t_{loiter}	1800	s
Specific fuel consumption (acc. to PL and OE)	SFC	**2,41E-05**	kg/N/s
Specific fuel consumption (acc. to fuel capacity)	SFC	**2,33E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	44,42	m
Structural wing span	$b_{W,struct}$	54,23	m
Wing area	S_W	283,4	m²
Aspect ratio	A	6,96	
Sweep	φ_{25}	35	°
Mean aerodynamic chord	c_{MAC}	7,36	m
Position of maximum camber	$x_{(y\_c),max}$	15	%c
Camber	$(y_c)_{max}/c$	4	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	10,0	%
Taper	λ	0,259	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	74,28	m/s
Mach number, landing	M_{APP}	0,22	
Mach number, cruise	M_{CR}	0,82	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,6	%c
Leading edge sweep	φ_{LE}	39,8	°
Reynoldsnumber	Re	3,9E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,47	
Correction term, camber	$\Delta_1 c_{L,max}$	0,27	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,022	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,761	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,73	
Correction term, Mach number	$\Delta C_{L,max}$	-0,01	
Lift coefficient, wing	$C_{L,max}$	1,28	

Calculations increase of lift coefficient due to flaps		**2 flap types**
Correction factor, sweep	K_φ	0,81
• Flap group A		
Double-slotted flap	$\Delta c_{L,max,fA}$	1,25
Use flapped span	b_W,fA	**19** m
Percentage of flaps allong the wing		35%
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,36
• Flap group B		
0,3c Single-slotted fowler flap	$\Delta c_{L,max,fB}$	1,52
Use flapped span	b_W,fB	**3,8** m
Percentage of flaps allong the wing		7%
Increase in maximum lift coefficient, flap group B	$\Delta C_{L,max,fB}$	0,09
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,44

Calculations increase of lift coefficient due to slats		**1 slat type**
Sweep angle of the hinge line	$\varphi_{H.L.}$	**38** °
• Slat group A		
Handley Page slat	$\Delta c_{L,max,sA}$	0,55
Use slatted span	b_W,sA	**41,8** m
Percentage of slats allong the wing		77%
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,33
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,33

Wing		
Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,03**
RE value maximum lift coefficient, landing		1,94
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,89**
RE value maximum lift coefficient, take-off		1,80
		5%

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
No winglets	$k_{e,WL}$	1,00
Aspect ratio	A	6,96
Effective aspect ratio	A_{eff}	6,96
Efficiency factor, short range	k_E	16,19
Relative wetted area	S_{wet}/S_W	**6,20**
Verification value maximum aerodynamic efficiency	E_{max}	**17,2**
RE value maximum aerodynamic efficiency		17,90
-4%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,820	
Cruise altitude	h_{CR}	10668	m
By Pass Ratio	μ	1,43	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	84,50	kN
Overall Pressure ratio	OAPR	**13,50**	
Turbine entry temperature	TET	**1425,33**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,97	
Ventilator efficiency	$\eta_{ventilator}$	0,78	
Compressor efficiency	$\eta_{compresor}$	0,80	
Turbine efficiency	$\eta_{turbine}$	0,88	
Nozzle efficiency	η_{nozzle}	0,98	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	218,81	K
Dimensionless turbine entry temperature	ϕ	6,51	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,13	
Temperature function	χ	1,25	
Gas generator efficiency	η_{gasgen}	0,99	
Gas generator function	G	2,05	
Verification value specific fuel consumption	SFC	0,86	kg/daN/h
Verification value specific fuel consumption	SFC	**2,38E-05**	kg/N/s
RE value specific fuel consumption (acc. to PL and OE)	SFC	2,41E-05	kg/N/s
-2%			
RE value specific fuel consumption (acc. to fuel capacity)	SFC	2,33E-05	kg/N/s
2%			

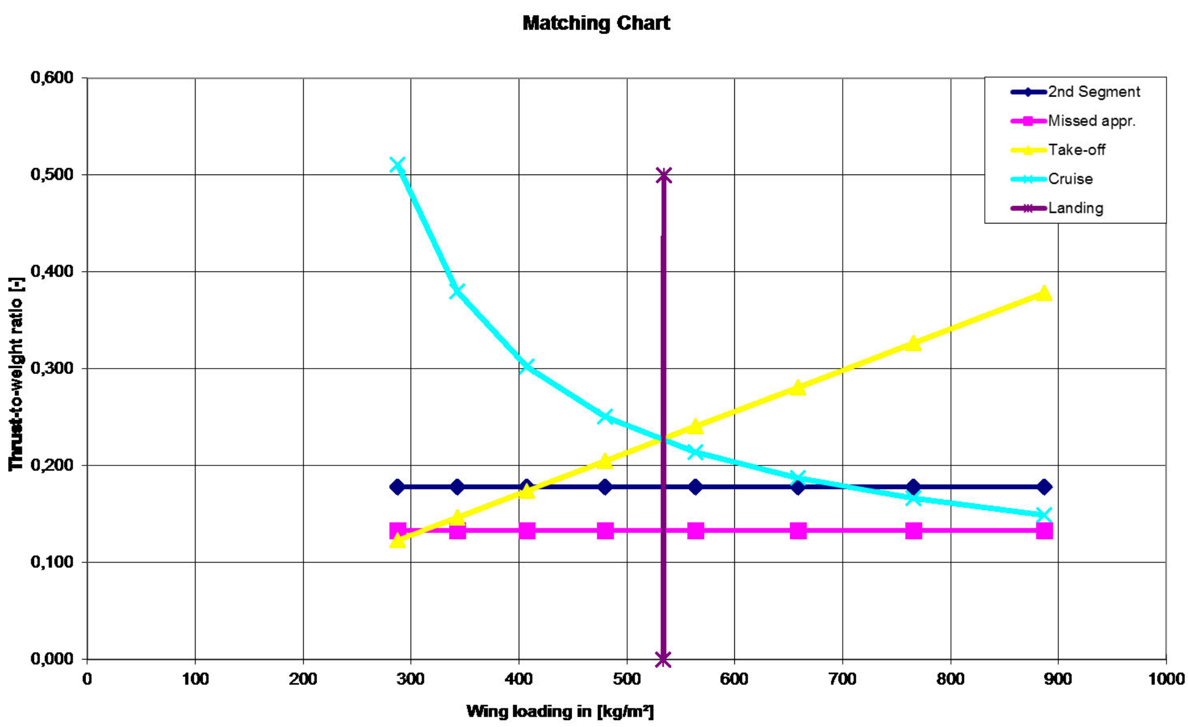

Figure B.2 Matching chart – Boeing 707-320C (minimum drag speed)

Appendix C BAe 146-200 (British Aerospace)

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**1173**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	58,3	58,3
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**1564**	m	1564	1564
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Range (maximum payload)		R	**1000**	NM		
Cruise Mach number		M_{CR}	**0,73**			
Wing area		S_W	**77**	m²		
Wing span	**Known**	b_W	**26,34**	m²	26,34	26,34
Aspect ratio		A	8,98			
Maximum take-off mass		m_{MTO}	**40597**	kg		
Maximum payload mass		m_{PL}	**10206**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,251			
Maximum landing mass		m_{ML}	**35154**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,866			
Operating empty mass		m_{OE}	**22000**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,542			
Wing loading		m_{MTO}/S_W	525,2	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**29,8**	kN		
Total take-off thrust		T_{TO}	119,2	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,299			
Bypass ratio		μ	**5,65**			

Data to optimize V/V_{md}

					LL	UL
Cruise speed		V_{CR}	**197**	m/s		
Cruise altitude		h_{CR}	**9144**	m		
Speed ratio		V/V_{md}	**1,316**	-	1,000	1,316

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**15**	°		
Mean aerodynamic chord		c_{MAC}	**3,17**	m		
Position of maximum camber		$x_{(y\_c),max}$	**50**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**6**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**13,0**	%		
Taper		λ	**0,356**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1173	1173	m	0,00%
Approach speed	V_{APP}	Unknown	**58,3**	m/s	0,00%
Take-off field length	s_{TOFL}	1564	1564	m	0,00%
Span	b_W	26,34	26,34	m	0,00%
Aspect ratio	A	8,98	8,98		0,00%
Cruise speed	V_{CR}	197,0	220	m/s	11,83%
Cruise altitude	h_{CR}	9144	9473	m	3,60%
Squared Sum					**1,53E-02**
Absolute maximum deviation					11,8%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**3,62**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,63**	
Maximum aerodynamic efficiency	E_{max}	**14,51**	
Specific fuel consumption	SFC	**1,95E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1173	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	58,29	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,87	
Wing loading at maximum take-off mass	m_{MTO}/S_W	525,2	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**3,62**	

Take-off

Take-off field length	s_{TOFL}	1564	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,299	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,63**	

2nd Segment

Aspect ratio	A	8,975
Lift coefficient, take-off	$C_{L,TO}$	1,82
Lift-independent drag coefficient, clean	$C_{D,0}$ (2^{nd} Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,036
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,056
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	8,12
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,204

Missed approach

Lift coefficient, landing	$C_{L,L}$	2,14
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,052
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,087
Aerodynamic efficiency in landing configuration	E_L	6,70
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,204

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,73	
Aspect ratio	A	8,98	
Bypass ratio	μ	5,65	
Wing loading	m_{MTO}/S_W	525	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,299	

Variables

	V/V_{md}	**1,316**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,028
Lift coefficient at E_{max}	$C_{L,md}$	0,83
Ratio, lift coefficient	$C_L/C_{L,md}$	0,577
Lift coefficient, cruise	C_L	0,477
Actual aerodynamic efficiency, cruise	E	12,57
Max. aerodynamic efficiency, cruise	E_{max}	**14,51**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	-0,19	0,00	0,00
f'(x)	-0,13	-0,12	-0,12
E_{max}	16	14,55	14,51

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	1000	NM
Mach number, cruise	M_{CR}	0,73	
Bypass ratio	μ	5,65	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,299	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	40597	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,251	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,542	

Calculated values			
Actual aerodynamic efficiency, cruise	E	12,57	
Cruise altitude	h_{CR}	9473	m
Cruise speed	V_{CR}	220	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	**0,998**	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,207	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,793	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	370400	m
Loiter time	t_{loiter}	2700	s
Specific fuel consumption	SFC	**1,95E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 63 series**	
Wing span	b_W	26,34	m
Structural wing span	$b_{W,struct}$	27,27	m
Wing area	S_W	77,3	m²
Aspect ratio	A	8,98	
Sweep	φ_{25}	15	°
Mean aerodynamic chord	c_{MAC}	3,17	m
Position of maximum camber	$x_{(y\_c),max}$	50	%c
Camber	$(y_c)_{max}/c$	6	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	13,0	%
Taper	λ	0,356	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	58,29	m/s
Mach number, landing	M_{APP}	0,18	
Mach number, cruise	M_{CR}	0,73	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,9	%c
Leading edge sweep	φ_{LE}	18,0	°
Reynoldsnumber	Re	1,3E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,54	
Correction term, camber	$\Delta_1 c_{L,max}$	0,34	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,020	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,904	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,84	
Correction term, Mach number	$\Delta C_{L,max}$	0,00	
Lift coefficient, wing	$C_{L,max}$	1,60	

Calculations increase of lift coefficient due to flaps		**1 flap type**	
Correction factor, sweep	K_φ	0,90	
• Flap group A			
0,4c Single-slotted fowler flap	$\Delta c_{L,max,fA}$	2,24	
Use flapped span	b_W,fA	**21,27**	m
Percentage of flaps allong the wing		78%	
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	1,57	
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	1,57	

Calculations increase of lift coefficient due to slats		**No slats**	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,00	

Wing

Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**3,10**
RE value maximum lift coefficient, landing		3,62
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,25**
RE value maximum lift coefficient, take-off		2,63
-14%		

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
No winglets	$k_{e,WL}$	1,00
Aspect ratio	A	8,98
Effective aspect ratio	A_{eff}	8,98
Efficiency factor, short range	k_E	15,15
Relative wetted area	S_{wet}/S_W	**6,10**
Verification value maximum aerodynamic efficiency	E_{max}	**18,4**
RE value maximum aerodynamic efficiency		14,51
27%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,730	
Cruise altitude	h_{CR}	9144	m
By Pass Ratio	μ	5,65	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	29,80	kN
Overall Pressure ratio	OAPR	**12,20**	
Turbine entry temperature	TET	**1251,54**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,95	
Ventilator efficiency	$\eta_{ventilator}$	0,76	
Compressor efficiency	$\eta_{compresor}$	0,82	
Turbine efficiency	$\eta_{turbine}$	0,83	
Nozzle efficiency	η_{nozzle}	0,94	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	228,71	K
Dimensionless turbine entry temperature	ϕ	5,47	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,11	
Temperature function	χ	1,15	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	1,42	
Verification value specific fuel consumption	SFC	0,92	kg/daN/h
Verification value specific fuel consumption	SFC	**2,55E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,95E-05	kg/N/s
	31%		

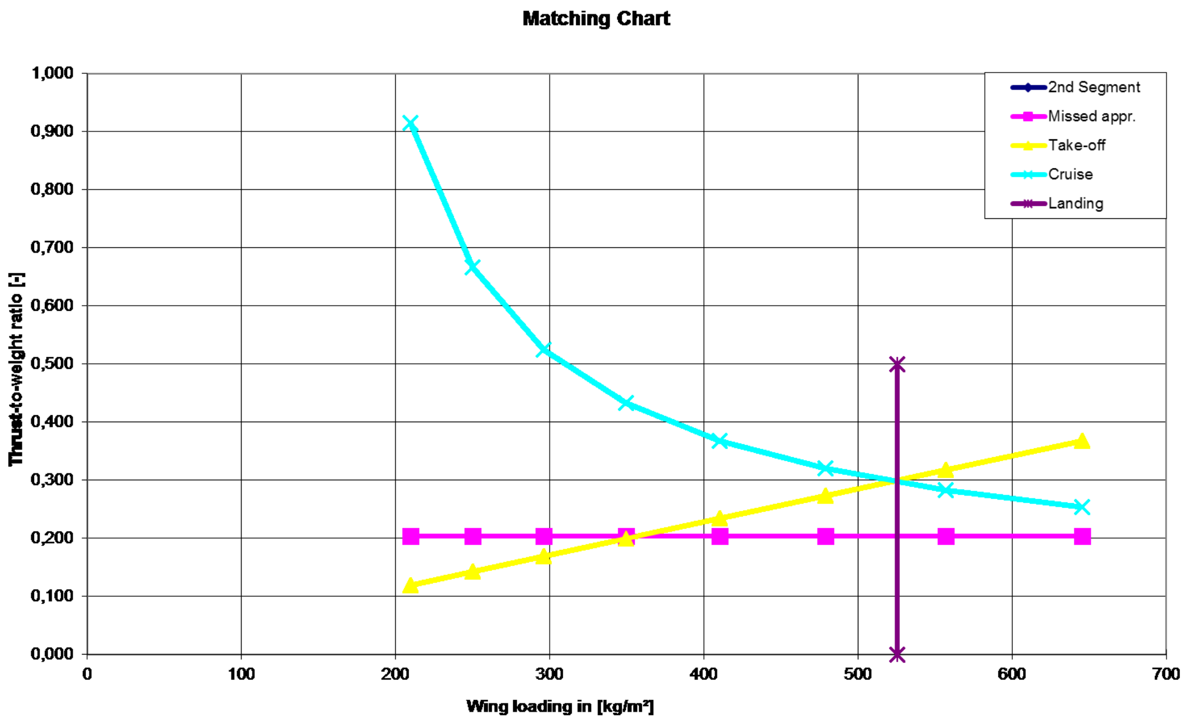

Figure C.1 Matching chart – BAe 146-200

Appendix D A320-200 (Airbus)

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**1700**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	70,2	70,2
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2200**	m	2200	2200
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Range (maximum payload)		R	**1600**	NM		
Cruise Mach number		M_{CR}	**0,78**			
Wing area		S_W	**122**	m²		
Wing span	**Known**	b_W	**34,09**	m²	34,09	34,09
Aspect ratio		A	9,50			
Maximum take-off mass		m_{MTO}	**73500**	kg		
Maximum payload mass		m_{PL}	**19000**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,259			
Maximum landing mass		m_{ML}	**64500**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,878			
Operating empty mass		m_{OE}	**41310**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,562			
Wing loading		m_{MTO}/S_W	600,8	kg/m²		
Number of engines		n_E	**2**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**111,2**	kN		
Total take-off thrust		T_{TO}	222,4	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,308			
Bypass ratio		μ	**6**			

Data to optimize V/V_{md}

				LL	UL
Cruise speed	V_{CR}	**230**	m/s		
Cruise altitude	h_{CR}	**11280**	m		
Speed ratio	V/V_{md}	**1,000**	-	1	**1,0**

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**25**	°		
Mean aerodynamic chord		c_{MAC}	**4,29**	m		
Position of maximum camber		$x_{(y\_c),max}$	**30**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**4**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**12,0**	%		
Taper		λ	**0,24**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1700	1700	m	0,00%
Approach speed	V_{APP}	Unknown	**70,2**	m/s	0,00%
Take-off field length	s_{TOFL}	2200	2200	m	0,00%
Span	b_W	34,09	34,09	m	0,00%
Aspect ratio	A	9,50	9,50		0,00%
Cruise speed	V_{CR}	230,0	230	m/s	0,08%
Cruise altitude	h_{CR}	11280	11995	m	6,34%
Squared Sum					**4,02E-03**
Absolute maximum deviation					6,3%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**2,90**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,07**	
Maximum aerodynamic efficiency	E_{max}	**17,91**	
Specific fuel consumption	SFC	**1,62E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1700	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	70,17	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,88	
Wing loading at maximum take-off mass	m_{MTO}/S_W	600,8	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**2,90**	

Take-off

Take-off field length	s_{TOFL}	2200	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,308	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,07**	

2nd Segment

Aspect ratio	A	9,500
Lift coefficient, take-off	$C_{L,TO}$	1,44
Lift-independent drag coefficient, clean	$C_{D,0}$ (2^{nd} Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,017
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,037
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	10,58
Number of engines	n_E	2
Climb gradient	$\sin(\gamma)$	0,024
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,237

Missed approach

Lift coefficient, landing	$C_{L,L}$	1,72
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,031
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,066
Aerodynamic efficiency in landing configuration	E_L	8,30
Climb gradient	$\sin(\gamma)$	0,021
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,248

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,78	
Aspect ratio	A	9,50	
Bypass ratio	μ	6,00	
Wing loading	m_{MTO}/S_W	601	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,308	

Variables

	V/V_{md}	**1,0**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,020
Lift coefficient at E_{max}	$C_{L,md}$	0,71
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	0,708
Actual aerodynamic efficiency, cruise	E	17,91
Max. aerodynamic efficiency, cruise	E_{max}	**17,91**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,21	-0,01	0,00
f'(x)	-0,11	-0,12	-0,12
E_{max}	16	17,98	17,91

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	1600	NM
Mach number, cruise	M_{CR}	0,78	
Bypass ratio	μ	6,00	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,308	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	73500	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,259	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,562	

Calculated values			
Actual aerodynamic efficiency, cruise	E	17,91	
Cruise altitude	h_{CR}	11995	m
Cruise speed	V_{CR}	230	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	**0,999**	
Fuel-Fraction, taxi	$M_{ff,taxi}$	**0,996**	
Fuel-Fraction, take-off	$M_{ff,TO}$	**0,993**	
Fuel-Fraction, climb	$M_{ff,CLB}$	**0,993**	
Fuel-Fraction, descent	$M_{ff,DES}$	**0,992**	
Fuel-Fraction, landing	$M_{ff,L}$	**0,992**	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,179	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,821	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	370400	m
Loiter time	t_{loiter}	2700	s
Specific fuel consumption	SFC	**1,62E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	34,09	m
Structural wing span	$b_{W,struct}$	37,61	m
Wing area	S_W	122,3	m²
Aspect ratio	A	9,50	
Sweep	φ_{25}	25	°
Mean aerodynamic chord	c_{MAC}	4,29	m
Position of maximum camber	$x_{(y\_c),max}$	30	%c
Camber	$(y_c)_{max}/c$	4	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	12,0	%
Taper	λ	0,24	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	70,17	m/s
Mach number, landing	M_{APP}	0,21	
Mach number, cruise	M_{CR}	0,78	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	3,1	%c
Leading edge sweep	φ_{LE}	28,7	°
Reynoldsnumber	Re	2,1E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,59	
Correction term, camber	$\Delta_1 c_{L,max}$	0,15	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,103	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,842	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,80	
Correction term, Mach number	$\Delta C_{L,max}$	-0,01	
Lift coefficient, wing	$C_{L,max}$	1,46	

Calculations increase of lift coefficient due to flaps		**1 flap type**
Correction factor, sweep	K_{φ}	0,87
• Flap group A		
Single-slotted flap	$\Delta c_{L,max,fA}$	0,78
Use flapped span	b_W,fA	**23,35** m
Percentage of flaps allong the wing		62%
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,42
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,42
Calculations increase of lift coefficient due to slats		**1 slat type**
Sweep angle of the hinge line	$\varphi_{H.L.}$	**27** °
• Slat group A		
0,3c Nose flap	$\Delta c_{L,max,sA}$	0,90
Use slatted span	b_W,sA	**30,82** m
Percentage of slats allong the wing		82%
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,66
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,66
Wing		
Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,52**
RE value maximum lift coefficient, landing		2,90
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,80**
RE value maximum lift coefficient, take-off		2,07
-13%		

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
End plate	$k_{e,WL}$	1,12
Span	b_W	34,09 m
Winglet height	h	**2,7** m
Aspect ratio	A	9,50
Effective aspect ratio	A_{eff}	10,59
Efficiency factor, short range	k_E	15,15
Relative wetted area	S_{wet}/S_W	**6,35**
Verification value maximum aerodynamic efficiency	E_{max}	**19,6**
RE value maximum aerodynamic efficiency		17,91
9%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,780	
Cruise altitude	h_{CR}	11280	m
By Pass Ratio	μ	6,00	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	111,20	kN
Overall Pressure ratio	OAPR	**26,50**	
Turbine entry temperature	TET	**1448,06**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,94	
Ventilator efficiency	$\eta_{ventilator}$	0,87	
Compressor efficiency	$\eta_{compresor}$	0,86	
Turbine efficiency	$\eta_{turbine}$	0,90	
Nozzle efficiency	η_{nozzle}	0,98	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	216,65	K
Dimensionless turbine entry temperature	ϕ	6,68	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,12	
Temperature function	χ	1,74	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	2,17	
Verification value specific fuel consumption	SFC	0,59	kg/daN/h
Verification value specific fuel consumption	SFC	**1,63E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,62E-05	kg/N/s

1%

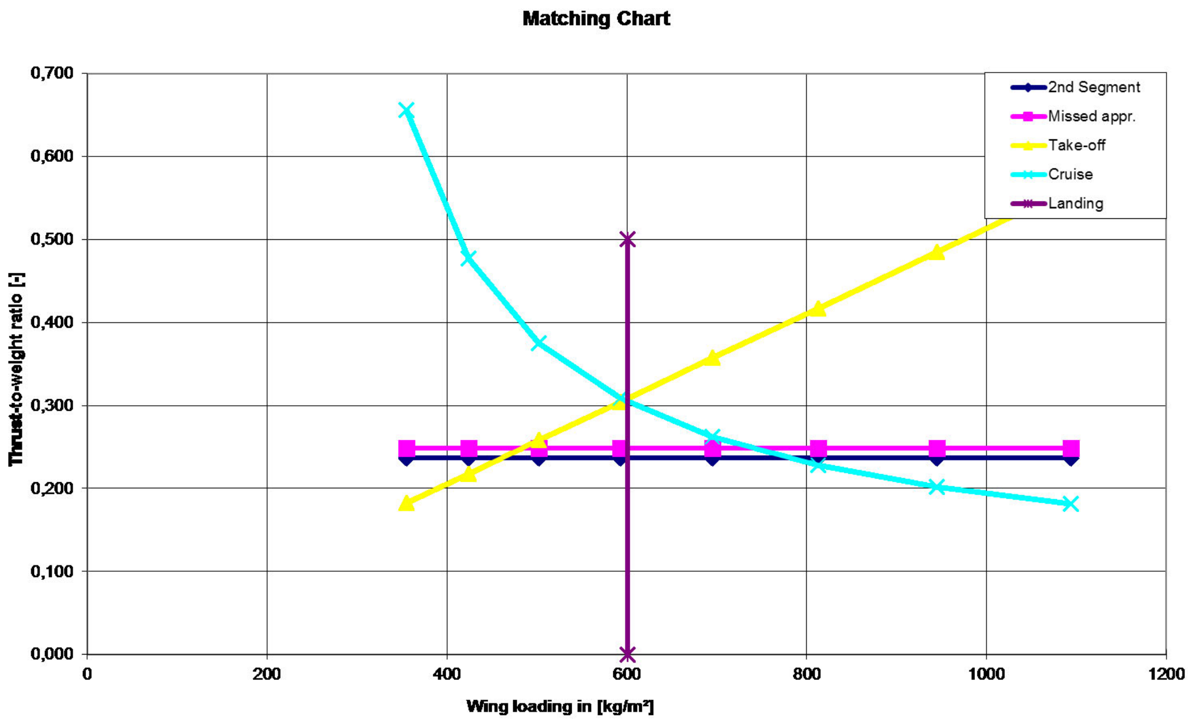

Figure D.1 Matching chart – A320-200

Appendix E The Rebel

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**2700**	m		
Approach speed	**Unknown**	V_{APP}	69,70	m/s	88,4	88,4
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2700**	m	2700	2700
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Range (acc. Payload-range diagram)		R	**1510**	NM		
Cruise Mach number		M_{CR}	**0,55**			
Wing area		S_W	68	m²		
Wing span	**Known**	b_W	**48,5**	m²	48,5	48,5
Aspect ratio		A	**34,80**			
Maximum take-off mass		m_{MTO}	**66000**	kg		
Payload mass (acc. Payload-range diagram)		m_{PL}	**19256**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,292			
Maximum landing mass		m_{ML}	60720	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	**0,920**			
Operating empty mass		m_{OE}	**39200**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,594			
Wing loading		m_{MTO}/S_W	976,4	kg/m²		
Number of engines		n_E	**2**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**89,1**	kN		
Total take-off thrust		T_{TO}	178,2	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,275			
Bypass ratio		μ	**15,5**			

Data to optimize V/V_{md}

				LL	UL
Cruise speed	V_{CR}	**167**	m/s		
Cruise altitude	h_{CR}	**9144**	m		
Speed ratio	V/V_{md}	**1,000**	-	1	**1,0**

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	2700	2700	m	0,00%
Approach speed	V_{APP}	Unknown	**88,4**	m/s	0,00%
Take-off field length	s_{TOFL}	2700	2700	m	0,00%
Span	b_W	48,50	48,50	m	0,00%
Aspect ratio	A	34,80	34,80		0,00%
Cruise speed	V_{CR}	166,8	166	m/s	-0,18%
Cruise altitude	h_{CR}	9144	9270	m	1,38%
Squared Sum					**1,94E-04**
Absolute maximum deviation					1,4%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**3,11**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**3,07**	
Maximum aerodynamic efficiency	E_{max}	**24,74**	
Specific fuel consumption	SFC	**1,03E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing			
Landing field length	s_{LFL}	2700	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	88,43	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,92	
Wing loading at maximum take-off mass	m_{MTO}/S_W	976,4	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**3,11**	

Take-off			
Take-off field length	s_{TOFL}	2700	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,275	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**3,07**	

2nd Segment		
Aspect ratio	A	34,800
Lift coefficient, take-off	$C_{L,TO}$	2,14
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,052
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,072
Oswald efficiency factor; landing configuration	e	**0,490**
Aerodynamic efficiency in take-off configuration	E_{TO}	13,61
Number of engines	n_E	2
Climb gradient	$\sin(\gamma)$	0,024
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,195

Missed approach		
Lift coefficient, landing	$C_{L,L}$	1,84
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,037
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,072
Aerodynamic efficiency in landing configuration	E_L	13,60
Climb gradient	$\sin(\gamma)$	0,021
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,174

2) Maximum Aerodynamic Efficiency

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	**0,68**	

Specifications			
Mach number, cruise	M_{CR}	0,55	
Aspect ratio	A	34,80	
Bypass ratio	μ	15,50	
Wing loading	m_{MTO}/S_W	976	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,275	

Variables		
	V/V_{md}	**1,0**

Calculations		
Zero-lift drag coefficient	$C_{D,0}$	0,030
Lift coefficient at E_{max}	$C_{L,md}$	1,50
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	1,503
Actual aerodynamic efficiency, cruise	E	24,74
Max. aerodynamic efficiency, cruise	E_{max}	**24,74**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,59	-0,11	0,00
f'(x)	-0,06	-0,08	-0,08
E_{max}	16	26,16	24,76

3) Specific Fuel Consumption

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications

Range	R	1510	NM
Mach number, cruise	M_{CR}	0,55	
Bypass ratio	μ	15,50	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,275	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	66000	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,292	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,594	

Calculated values

Actual aerodynamic efficiency, cruise	E	24,74	
Cruise altitude	h_{CR}	9270	m
Cruise speed	V_{CR}	166	m/s

Mission fuel fraction

Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	**1,000**	
Fuel-Fraction, taxi	$M_{ff,taxi}$	**0,997**	
Fuel-Fraction, take-off	$M_{ff,TO}$	**0,994**	
Fuel-Fraction, climb	$M_{ff,CLB}$	**0,994**	
Fuel-Fraction, descent	$M_{ff,DES}$	**0,994**	
Fuel-Fraction, landing	$M_{ff,L}$	**0,994**	

Calculations

Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,114	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,886	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	370400	m
Loiter time	t_{loiter}	**1920**	s
Specific fuel consumption	SFC	**1,03E-05**	kg/N/s

4) Verification Specifications

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83	
No winglets	$k_{e,WL}$	1,00	
Aspect ratio	A	34,80	
Effective aspect ratio	A_{eff}	34,80	
Efficiency factor, short range	k_E	**13,3**	
Relative wetted area	S_{wet}/S_W	**9,10**	
Verification value maximum aerodynamic efficiency	E_{max}	**26,1**	
RE value maximum aerodynamic efficiency		24,74	

5%

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,550	
Cruise altitude	h_{CR}	9144	m
By Pass Ratio	μ	15,50	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	89,10	kN
Overall Pressure ratio	OAPR	**54,55**	
Turbine entry temperature	TET	**1430,21**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,90	
Ventilator efficiency	$\eta_{ventilator}$	0,90	
Compressor efficiency	$\eta_{compresor}$	0,88	
Turbine efficiency	$\eta_{turbine}$	0,93	
Nozzle efficiency	η_{nozzle}	0,98	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	$T(H)$	228,71	K
Dimensionless turbine entry temperature	ϕ	6,25	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,06	
Temperature function	χ	2,26	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	1,66	
Verification value specific fuel consumption	SFC	0,37	kg/daN/h
Verification value specific fuel consumption	SFC	**1,03E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,03E-05	kg/N/s

1%

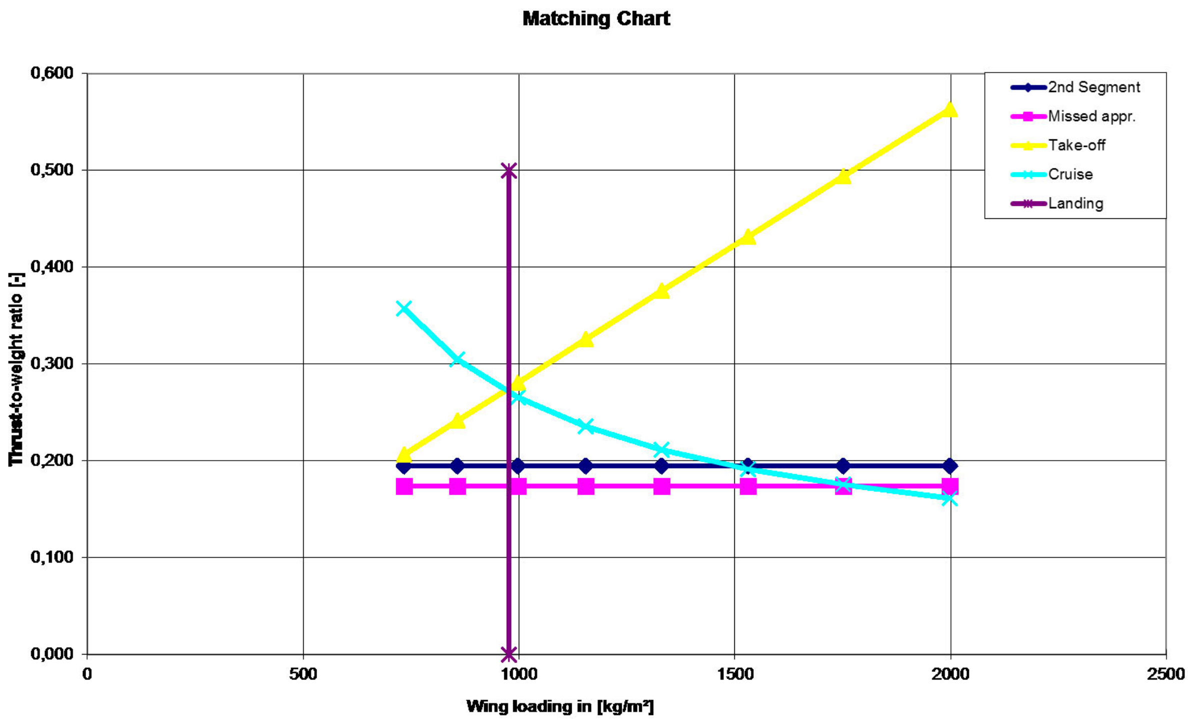

Figure E.1 Matching chart – The Rebel

Appendix F Boeing SUGAR High

Aeroplane Specifications

Data to apply reverse engineering

					LL	*UL*
Landing field length	**Unknown**	s_{LFL}	1583	m		
Approach speed	**Known**	V_{APP}	**59,16**	m/s	59,2	59,2
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2496**	m	2496	2496
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Range (acc. Payload-range diagram)		R	**3500**	NM		
Cruise Mach number		M_{CR}	**0,74**			
Wing area		S_W	**158**	m²		
Wing span	**Known**	b_W	61,6	m²	61,5	61,5
Aspect ratio		A	**24,00**			
Maximum take-off mass		m_{MTO}	**74322**	kg		
Payload mass (acc. Payload-range diagram)		m_{PL}	**16416**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,221			
Maximum landing mass		m_{ML}	70329	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	**0,946**			
Operating empty mass		m_{OE}	**48484**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,652			
Wing loading		m_{MTO}/S_W	470,6	kg/m²		
Number of engines		n_E	**2**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**84,1**	kN		
Total take-off thrust		T_{TO}	168	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,231			
Bypass ratio		μ	**9,2**			

Data to optimize V/V_{md}

				LL	*UL*
Cruise speed	V_{CR}	**218**	m/s		
Cruise altitude	h_{CR}	**13411**	m		
Speed ratio	V/V_{md}	**1,316**	-	**1,316**	1,32

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	Unknown	**1056**	m	0,00%
Approach speed	V_{APP}	59,16	59,2	m/s	0,00%
Take-off field length	s_{TOFL}	2496,312	2496	m	0,00%
Span	b_W	61,5665821	61,566582 1	m	0,00%
Aspect ratio	A	24,00	24,00		0,00%
Cruise speed	V_{CR}	218,4	218	m/s	0,00%
Cruise altitude	h_{CR}	13411	11472	m	-14,46%
Squared Sum					**2,09E-02**
Absolute maximum deviation					14,5%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**3,44**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,91**	
Maximum aerodynamic efficiency	E_{max}	**30,14**	
Specific fuel consumption	SFC	**6,82E-06**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1056	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	**1,82**	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	59,16	m/s
Factor, landing	k_L	**0,122**	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,95	
Wing loading at maximum take-off mass	m_{MTO}/S_W	470,6	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**3,44**	

Take-off

Take-off field length	s_{TOFL}	2496	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,231	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,91**	

2nd Segment

Aspect ratio	A	24,000
Lift coefficient, take-off	$C_{L,TO}$	1,33
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,011
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,031
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	20,48
Number of engines	n_E	2
Climb gradient	$\sin(\gamma)$	0,024
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,146

Missed approach

Lift coefficient, landing	$C_{L,L}$	2,04
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,047
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,082
Aerodynamic efficiency in landing configuration	E_L	12,69
Climb gradient	$\sin(\gamma)$	0,021
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,189

2) Maximum Aerodynamic Efficiency

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	**0,783**	

Specifications			
Mach number, cruise	M_{CR}	0,74	
Aspect ratio	A	24,00	
Bypass ratio	μ	9,20	
Wing loading	m_{MTO}/S_W	471	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,231	

Variables		
	V/V_{md}	**1,3**

Calculations		
Zero-lift drag coefficient	$C_{D,0}$	0,016
Lift coefficient at E_{max}	$C_{L,md}$	0,98
Ratio, lift coefficient	$C_L/C_{L,md}$	0,577
Lift coefficient, cruise	C_L	0,565
Actual aerodynamic efficiency, cruise	E	26,10
Max. aerodynamic efficiency, cruise	E_{max}	**30,14**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,80	-0,31	-0,01
f'(x)	-0,04	-0,07	-0,07
E_{max}	16	34,52	30,31

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	3500	NM
Mach number, cruise	M_{CR}	0,74	
Bypass ratio	μ	9,20	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,231	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	74322,47552	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,221	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,652	

Calculated values			
Actual aerodynamic efficiency, cruise	E	26,10	
Cruise altitude	h_{CR}	11472	m
Cruise speed	V_{CR}	218	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	**1,000**	
Fuel-Fraction, taxi	$M_{ff,taxi}$	**0,997**	
Fuel-Fraction, take-off	$M_{ff,TO}$	**0,992**	
Fuel-Fraction, climb	$M_{ff,CLB}$	**0,992**	
Fuel-Fraction, descent	$M_{ff,DES}$	**0,992**	
Fuel-Fraction, landing	$M_{ff,L}$	**0,992**	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,127	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,873	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	370400	m
Loiter time	t_{loiter}	2700	s
Specific fuel consumption	SFC	**6,82E-06**	kg/N/s

4) Verification Specifications

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83
No winglets	$k_{e,WL}$	1,00
Aspect ratio	A	24,00
Effective aspect ratio	A_{eff}	24,00
Efficiency factor, short range	k_E	16,19
Relative wetted area	S_{wet}/S_W	**6,10**
Verification value maximum aerodynamic efficiency	E_{max}	**32,1**
RE value maximum aerodynamic efficiency		30,14
7%		

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,740	
Cruise altitude	h_{CR}	13411	m
By Pass Ratio	μ	9,20	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	84,07	kN
Overall Pressure ratio	OAPR	**32,40**	
Turbine entry temperature	TET	**1424,84**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,93	
Ventilator efficiency	$\eta_{ventilator}$	0,86	
Compressor efficiency	$\eta_{compresor}$	0,86	
Turbine efficiency	$\eta_{turbine}$	0,89	
Nozzle efficiency	η_{nozzle}	0,98	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	$T(H)$	216,65	K
Dimensionless turbine entry temperature	ϕ	6,58	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,11	
Temperature function	χ	1,89	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	2,02	

Verification value specific fuel consumption	SFC	0,55	kg/daN/h
Verification value specific fuel consumption	SFC	**1,52E-05**	kg/N/s
RE value specific fuel consumption	SFC	6,82E-06	kg/N/s
123%			
42% of the specific fuel consumption	$SFC_{42\%}$	**6,39E-06**	kg/N/s
-7%			

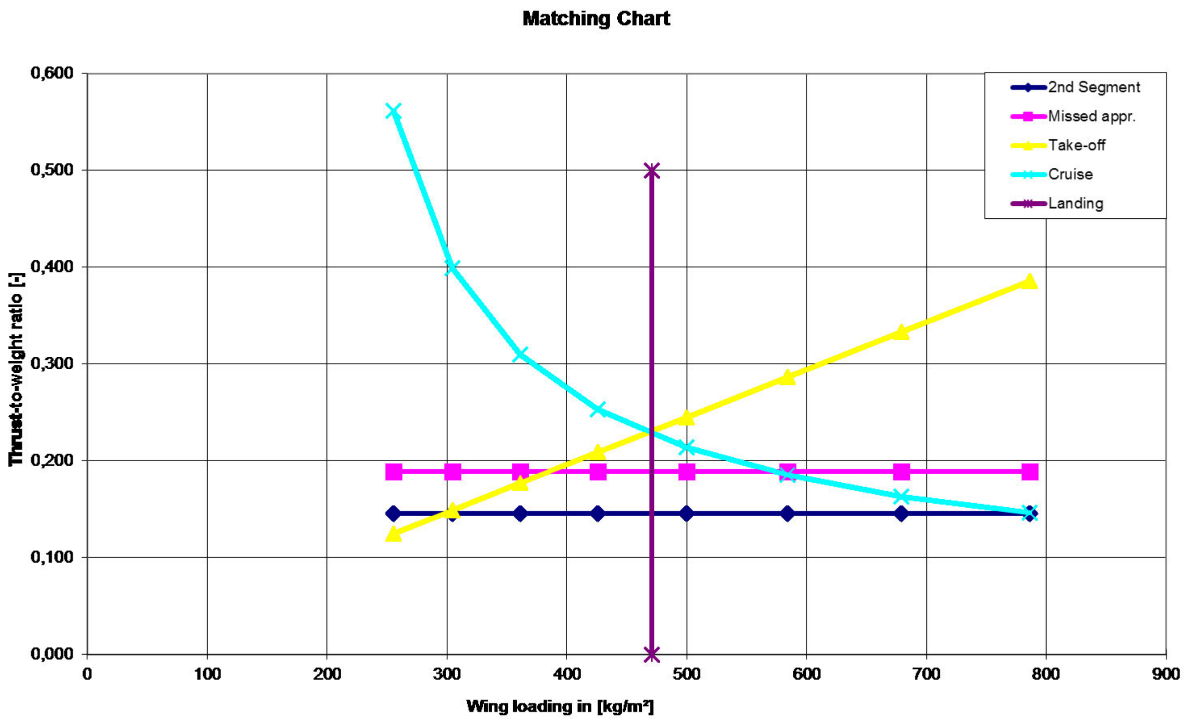

Figure F.1 Matching chart – Boeing SUGAR High

Appendix G Boeing 747-400

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Known**	s_{LFL}	**1905**	m		
Approach speed	**Known**	V_{APP}	**75,10**	m/s	75,1	75,1
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**2815**	m	2815	2815
Temperature above ISA (288,15K)		ΔT_{TO}	**15**	K		
Relative density		σ	**0,951**			
Range (maximum payload)		R	**4890**	NM		
Cruise Mach number		M_{CR}	**0,855**			
Wing area		S_W	**541**	m²		
Wing span	**Known**	b_W	**64,44**	m²	64,44	64,44
Aspect ratio		A	7,67			
Maximum take-off mass		m_{MTO}	**362870**	kg		
Maximum payload mass		m_{PL}	**39728**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,109			
Maximum landing mass		m_{ML}	**260360**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,718			
Operating empty mass		m_{OE}	**183160**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,505			
Wing loading		m_{MTO}/S_W	670,5	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**253**	kN		
Total take-off thrust		T_{TO}	1012	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,284			
Bypass ratio		μ	**4,85**			

Data to optimize V/V_{md}

					LL	UL
Cruise speed		V_{CR}	**254**	m/s		
Cruise altitude		h_{CR}	**10622**	m		
Speed ratio		V/V_{md}	**1,000**	-	1	**1,0**

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**37,5**	°		
Mean aerodynamic chord		c_{MAC}	**9,68**	m		
Position of maximum camber		$x_{(y\_c),max}$	**30**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**6**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**35**	%c	30 - 45	%c
Relative thickness	**Known**	t/c	**9,4**	%		
Taper		λ	**0,275**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	1905	1905	m	0,00%
Approach speed	V_{APP}	75,10	75,1	m/s	0,00%
Take-off field length	s_{TOFL}	2815	2815	m	0,00%
Span	b_W	64,44	64,44	m	0,00%
Aspect ratio	A	7,67	7,67		0,00%
Cruise speed	V_{CR}	254,3	252	m/s	-0,76%
Cruise altitude	h_{CR}	10622	11492	m	8,19%
Squared Sum					**6,77E-03**
Absolute maximum deviation					8,2%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**2,36**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,06**	
Maximum aerodynamic efficiency	E_{max}	**16,88**	
Specific fuel consumption	SFC	**1,74E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	1905	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	75,10	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,72	
Wing loading at maximum take-off mass	m_{MTO}/S_W	670,5	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**2,36**	

Take-off

Take-off field length	s_{TOFL}	2815	m
Temperatur above ISA (288,15K)	ΔT_{TO}	15	K
Relative density	σ	0,95	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,06**	

2nd Segment

Aspect ratio	A	7,673
Lift coefficient, take-off	$C_{L,TO}$	1,43
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,017
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,037
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	9,05
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,187

Missed approach

Lift coefficient, landing	$C_{L,L}$	1,40
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,015
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,050
Aerodynamic efficiency in landing configuration	E_L	8,44
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,139

2) Maximum Aerodynamic Efficiency

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications			
Mach number, cruise	M_{CR}	0,855	
Aspect ratio	A	7,67	
Bypass ratio	μ	4,85	
Wing loading	m_{MTO}/S_W	670	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	

Variables		
	V/V_{md}	**1,0**

Calculations		
Zero-lift drag coefficient	$C_{D,0}$	0,018
Lift coefficient at E_{max}	$C_{L,md}$	0,61
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	0,607
Actual aerodynamic efficiency, cruise	E	16,88
Max. aerodynamic efficiency, cruise	E_{max}	**16,88**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,10	0,00	0,00
f'(x)	-0,11	-0,12	-0,12
E_{max}	16	16,89	16,88

3) Specific Fuel Consumption

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications

Range	R	4890	NM
Mach number, cruise	M_{CR}	0,855	
Bypass ratio	μ	4,85	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,284	
Available fuel volume	$V_{fuel,available}$	204,35	m³
Maximum take-off mass	m_{MTO}	362870	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,109	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,505	

Calculated values

Actual aerodynamic efficiency, cruise	E	16,88	
Cruise altitude	h_{CR}	11492	m
Cruise speed	V_{CR}	252	m/s

Mission fuel fraction

Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations

Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,386	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,614	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**no**	
	international	**yes**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	823214	m
Loiter time	t_{loiter}	1800	s
Specific fuel consumption	SFC	**1,74E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	64,44	m
Structural wing span	$b_{W,struct}$	81,22	m
Wing area	S_W	541,2	m²
Aspect ratio	A	7,67	
Sweep	φ_{25}	37,5	°
Mean aerodynamic chord	c_{MAC}	9,68	m
Position of maximum camber	$x_{(y\_c),max}$	30	%c
Camber	$(y_c)_{max}/c$	6	%c
Position of maximum thickness	$x_{t,max}$	35	%c
Relative thickness	t/c	9,4	%
Taper	λ	0,275	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	75,10	m/s
Mach number, landing	M_{APP}	0,23	
Mach number, cruise	M_{CR}	0,855	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,4	%c
Leading edge sweep	φ_{LE}	41,7	°
Reynoldsnumber	Re	5,2E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,39	
Correction term, camber	$\Delta_1 c_{L,max}$	0,40	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,12	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,010	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,923	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,74	
Correction term, Mach number	$\Delta C_{L,max}$	-0,02	
Lift coefficient, wing	$C_{L,max}$	1,40	

Calculations increase of lift coefficient due to flaps			**1 flap type**
Correction factor, sweep	K_{φ}	0,80	
• Flap group A			
Double-slotted flap	$\Delta c_{L,max,fA}$	1,37	
Use flapped span	b_W,fA	**34,85**	m
Percentage of flaps allong the wing		43%	
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,47	
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,47	

Calculations increase of lift coefficient due to slats			**2 slat types**
Sweep angle of the hinge line	$\varphi_{H.L.}$	**42**	°
• Slat group A			
0,1c Kruger flap	$\Delta c_{L,max,sA}$	0,64	
Use slatted span	b_W,sA	**15,26**	m
Percentage of slats allong the wing		19%	
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,09	
• Slat group B			
0,3c Nose flap	$\Delta c_{L,max,SB}$	0,87	
Use slatted span	b_W,sB	**42,27**	m
Percentage of slats allong the wing		52%	
Increase in maximum lift coefficient, slat group B	$\Delta C_{L,max,sB}$	0,34	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,42	

Wing			
Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,27**	
RE value maximum lift coefficient, landing		2,36	
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,98**	
RE value maximum lift coefficient, take-off		2,06	
-4%			

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83	
End plate	$k_{e,WL}$	1,02	
Span	b_W	64,44	m
Winglet height	h	**0,89**	m
Aspect ratio	A	7,67	
Effective aspect ratio	A_{eff}	7,82	
Efficiency factor, short range	k_E	16,19	
Relative wetted area	S_{wet}/S_W	**6,30**	
Verification value maximum aerodynamic efficiency	E_{max}	**18,0**	
RE value maximum aerodynamic efficiency		16,88	
7%			

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,855	
Cruise altitude	h_{CR}	10622	m
By Pass Ratio	μ	4,85	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	253,00	kN
Overall Pressure ratio	OAPR	**30,20**	
Turbine entry temperature	TET	**1488,38**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,95	
Ventilator efficiency	$\eta_{ventilator}$	0,88	
Compressor efficiency	$\eta_{compresor}$	0,86	
Turbine efficiency	$\eta_{turbine}$	0,90	
Nozzle efficiency	η_{nozzle}	0,99	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	219,11	K
Dimensionless turbine entry temperature	ϕ	6,79	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,15	
Temperature function	χ	1,89	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	2,21	
Verification value specific fuel consumption	SFC	0,61	kg/daN/h
Verification value specific fuel consumption	SFC	**1,69E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,74E-05	kg/N/s

-3%

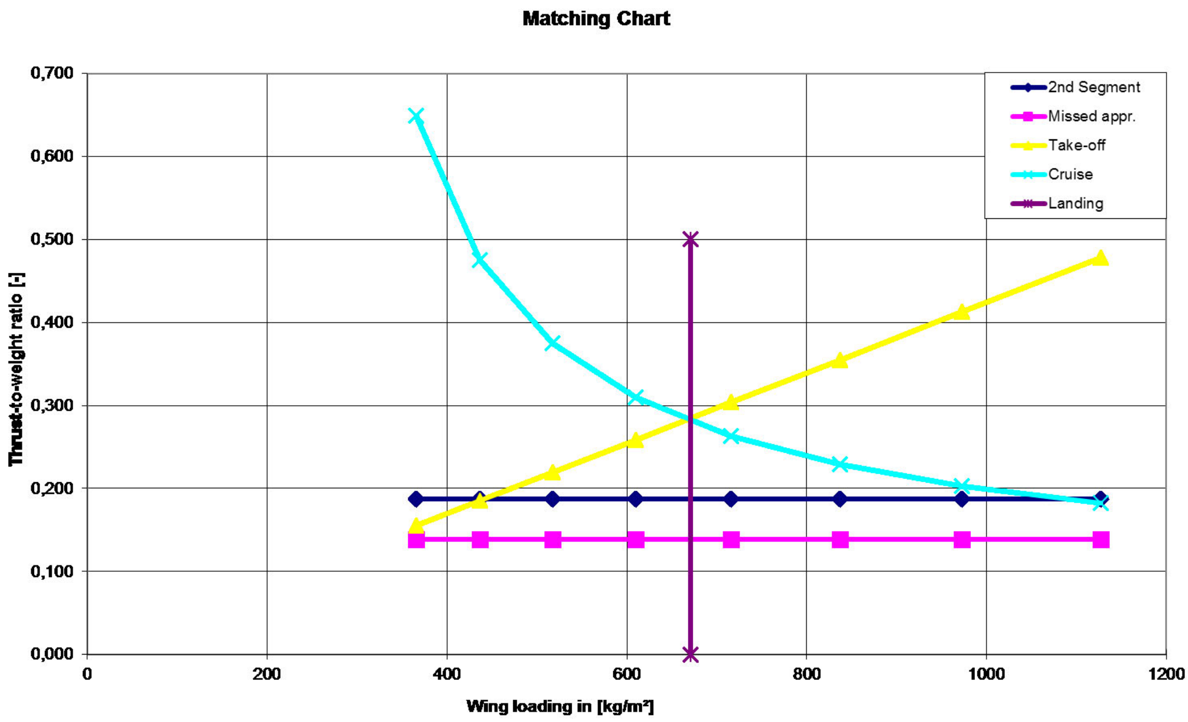

Figure G.1 Matching chart – Boeing 747-400

Appendix H Blended Wing Body VELA 2

Aeroplane Specifications

Data to apply reverse engineering

					LL	UL
Landing field length	**Unknown**	s_{LFL}	1583	m		
Approach speed	**Known**	V_{APP}	**84,88**	m/s	84,9	84,9
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**3350**	m	3350	3350
Temperature above ISA (288,15K)		ΔT_{TO}	**15**	K		
Relative density		σ	**0,951**			
Range (acc. Payload-range diagram)		R	**7500**	NM		
Cruise Mach number		M_{CR}	**0,85**			
Wing area		S_W	**1923**	m²		
Wing span	**Known**	b_W	**100,0**	m²	100	100
Aspect ratio		A	5,20			
Maximum take-off mass		m_{MTO}	**691200**	kg		
Payload mass (acc. Payload-range diagram)		m_{PL}	**83125**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,120			
Maximum landing mass		m_{ML}	366000	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	**0,530**			
Operating empty mass		m_{OE}	380600	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	**0,551**			
Wing loading		m_{MTO}/S_W	359,4	kg/m²		
Number of engines		n_E	**4**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**344**	kN		
Total take-off thrust		T_{TO}	1376	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,203			
Bypass ratio		μ	**9,5**			

Data to optimize V/V_{md}

				LL	UL
Cruise speed	V_{CR}	**251**	m/s		
Cruise altitude	h_{CR}	**10668**	m		
Speed ratio	V/V_{md}	**1,000**	-	1	1,316

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	Unknown	**2487**	m	0,00%
Approach speed	V_{APP}	84,88	84,9	m/s	0,00%
Take-off field length	s_{TOFL}	3350	3350	m	0,00%
Span	b_W	100	100	m	0,00%
Aspect ratio	A	5,20	5,20		0,00%
Cruise speed	V_{CR}	250,8	253	m/s	1,00%
Cruise altitude	h_{CR}	10668	10333	m	-3,14%
Squared Sum					**1,08E-03**
Absolute maximum deviation					3,1%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**0,72**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,30**	
Maximum aerodynamic efficiency	E_{max}	**25,36**	
Specific fuel consumption	SFC	**1,38E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	2487	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	1,70	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	84,88	m/s
Factor, landing	k_L	0,107	kg/m³
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,53	
Wing loading at maximum take-off mass	m_{MTO}/S_W	359,4	kg/m²
Maximum lift coefficient, landing	$C_{L,max,L}$	**0,72**	

Take-off

Take-off field length	s_{TOFL}	3350	m
Temperatur above ISA (288,15K)	ΔT_{TO}	15	K
Relative density	σ	0,95	
Factor	k_{TO}	2,34	m³/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,203	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,30**	

2nd Segment

Aspect ratio	A	5,200
Lift coefficient, take-off	$C_{L,TO}$	0,90
Lift-independent drag coefficient, clean	$C_{D,0}$ (2nd Segment)	**0,010**
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,000
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,010
Oswald efficiency factor; landing configuration	e	**0,8**
Aerodynamic efficiency in take-off configuration	E_{TO}	12,46
Number of engines	n_E	4
Climb gradient	$\sin(\gamma)$	0,030
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,147

Missed approach

Lift coefficient, landing	$C_{L,L}$	0,42
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,000
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,035
Aerodynamic efficiency in landing configuration	E_L	8,69
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,100

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,85	
Aspect ratio	A	5,20	
Bypass ratio	μ	9,50	
Wing loading	m_{MTO}/S_W	359	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,203	

Variables

	V/V_{md}	**1,0**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,005
Lift coefficient at E_{max}	$C_{L,md}$	0,27
Ratio, lift coefficient	$C_L/C_{L,md}$	1,000
Lift coefficient, cruise	C_L	0,274
Actual aerodynamic efficiency, cruise	E	25,36
Max. aerodynamic efficiency, cruise	E_{max}	**25,36**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,62	-0,13	0,00
f'(x)	-0,06	-0,08	-0,08
E_{max}	16	27,03	25,39

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	7500	NM
Mach number, cruise	M_{CR}	0,85	
Bypass ratio	μ	9,50	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,203	
Available fuel volume	$V_{fuel,available}$	23,86	m³
Maximum take-off mass	m_{MTO}	691200	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,120	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,551	

Calculated values			
Actual aerodynamic efficiency, cruise	E	25,36	
Cruise altitude	h_{CR}	10333	m
Cruise speed	V_{CR}	253	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Transport jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	0,990	
Fuel-Fraction, taxi	$M_{ff,taxi}$	0,990	
Fuel-Fraction, take-off	$M_{ff,TO}$	0,995	
Fuel-Fraction, climb	$M_{ff,CLB}$	0,980	
Fuel-Fraction, descent	$M_{ff,DES}$	0,990	
Fuel-Fraction, landing	$M_{ff,L}$	0,992	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,329	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,671	
Distance to alternate	$s_{to\_alternate}$	200	NM
Distance to alternate	$s_{to\_alternate}$	370400	m
Choose: FAR Part121-Reserves	domestic	**no**	
	international	**yes**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	1064900	m
Loiter time	t_{loiter}	1800	s
Specific fuel consumption	SFC	**1,38E-05**	kg/N/s

4) Verification Specifications

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,850	
Cruise altitude	h_{CR}	10668	m
By Pass Ratio	μ	9,50	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	344,00	kN
Overall Pressure ratio	OAPR	**33,46**	
Turbine entry temperature	TET	**1496,74**	
Inlet pressure loss	ΔP/P	2%	
Inlet efficiency	η_{inlet}	0,93	
Ventilator efficiency	$\eta_{ventilator}$	0,90	
Compressor efficiency	$\eta_{compresor}$	0,88	
Turbine efficiency	$\eta_{turbine}$	0,91	
Nozzle efficiency	η_{nozzle}	0,99	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	218,81	K
Dimensionless turbine entry temperature	ϕ	6,84	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,14	
Temperature function	χ	1,98	
Gas generator efficiency	η_{gasgen}	0,97	
Gas generator function	G	2,23	
Verification value specific fuel consumption	SFC	0,56	kg/daN/h
Verification value specific fuel consumption	SFC	**1,54E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,38E-05	kg/N/s

12%

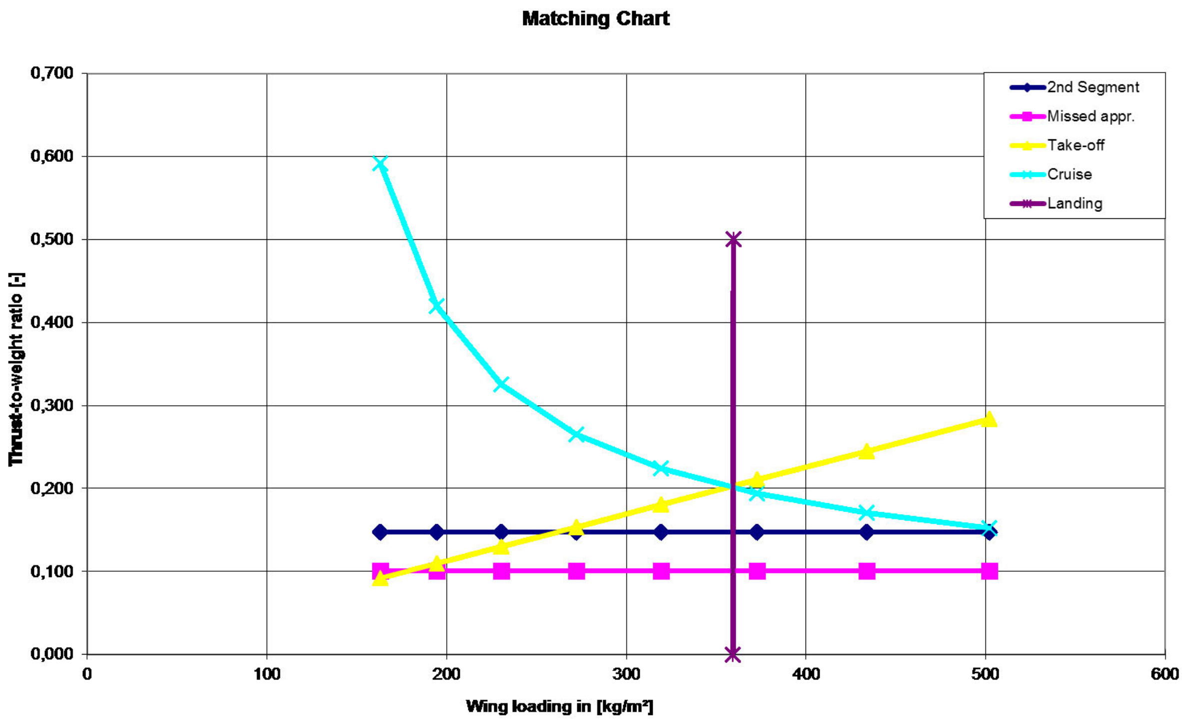

Figure H.1 Matching chart – Blended Wing Body VELA 2

Appendix I Dassault Falcon 8X

Aeroplane Specifications

Data to apply reverse engineering

					LL	*UL*
Landing field length	**Known**	s_{LFL}	**656**	m		
Approach speed	**Known**	V_{APP}	**54,53**	m/s	54,5	54,5
Temperature above ISA (288,15K)		ΔT_L	**0**	K		
Relative density		σ	**1**			
Take-off field length	**Known**	s_{TOFL}	**1829**	m	1829	1829
Temperature above ISA (288,15K)		ΔT_{TO}	**0**	K		
Relative density		σ	**1,000**			
Range (maximum payload)		R	**6450**	NM		
Cruise Mach number		M_{CR}	**0,8**			
Wing area		S_W	**71**	m²		
Wing span	**Known**	b_W	**26,29**	m²	26,29	26,29
Aspect ratio		A	9,78			
Maximum take-off mass		m_{MTO}	**33113**	kg		
Maximum payload mass		m_{PL}	**1800**	kg		
Mass ratio, payload - take-off		m_{PL}/m_{MTO}	0,054			
Maximum landing mass		m_{ML}	**29304**	kg		
Mass ratio, landing - take-off		m_{ML}/m_{MTO}	0,885			
Operating empty mass		m_{OE}	**16490,5**	kg		
Mass ratio, operating empty - take-off		m_{OE}/m_{MTO}	0,498			
Wing loading		m_{MTO}/S_W	468,4	kg/m²		
Number of engines		n_E	**3**			
Take-off thrust for one engine		$T_{TO,one\ engine}$	**29,9**	kN		
Total take-off thrust		T_{TO}	89,7	kN		
Thrust to weight ratio		$T_{TO}/(m_{MTO}*g)$	0,276			
Bypass ratio		μ	**4,5**			

Data to optimize V/V_{md}

				LL	UL
Cruise speed	V_{CR}	**236**	m/s		
Cruise altitude	h_{CR}	**11500**	m		
Speed ratio	V/V_{md}	**1,210**	-	1	1,316

Data to execute the verification

					Range	
Sweep angle		φ_{25}	**28,5**	°		
Mean aerodynamic chord		c_{MAC}	**3,8**	m		
Position of maximum camber		$x_{(y\_c),max}$	**25**	%c	15 - 50	%c
Camber		$(y_c)_{max}/c$	**4**	%c	2 - 6	%c
Position of maximum thickness		$x_{t,max}$	**30**	%c	30 - 45	%c
Relative thickness	**Unknown**	t/c	**11,3**	%		
Taper		λ	**0,22**			

Reverse Engineering

Reverse engineering & optimization of V/Vmd

	Quantity	*Original value*	RE value	*Unit*	*Deviation*
Landing field length	s_{LFL}	656	656	m	0,00%
Approach speed	V_{APP}	54,53	54,5	m/s	0,00%
Take-off field length	s_{TOFL}	1829	1829	m	0,00%
Span	b_W	26,29	26,29	m	0,00%
Aspect ratio	A	9,78	9,78		0,00%
Cruise speed	V_{CR}	236,1	236	m/s	-0,01%
Cruise altitude	h_{CR}	11500	11500	m	0,00%
Squared Sum					**5,67E-09**
Absolute maximum deviation					0,0%

Results reverse engineering

Maximum lift coefficient, landing	$C_{L,max,L}$	**3,70**	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,17**	
Maximum aerodynamic efficiency	E_{max}	**18,37**	
Specific fuel consumption	SFC	**1,80E-05**	kg/N/s

Reverse Engineering

1) Maximum Lift Coefficient for Landing and Take-off

Landing

Landing field length	s_{LFL}	656	m
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1,000	
Factor, approach	k_{APP}	**2,15**	$(m/s^2)^{0.5}$
Approach speed	V_{APP}	54,53	m/s
Factor, landing	k_L	**0,171**	kg/m^3
Mass ratio, landing - take-off	m_{ML}/m_{TO}	0,88	
Wing loading at maximum take-off mass	m_{MTO}/S_W	468,4	kg/m^2
Maximum lift coefficient, landing	$C_{L,max,L}$	**3,70**	

Take-off

Take-off field length	s_{TOFL}	1829	m
Temperatur above ISA (288,15K)	ΔT_{TO}	0	K
Relative density	σ	1,00	
Factor	k_{TO}	2,34	m^3/kg
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,276	
Maximum lift coefficient, take-off	$C_{L,max,TO}$	**2,17**	

2nd Segment

Aspect ratio	A	9,776
Lift coefficient, take-off	$C_{L,TO}$	1,51
Lift-independent drag coefficient, clean	$C_{D,0}$ (2^{nd} Segment)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,020
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Profile drag coefficient	$C_{D,P}$	0,040
Oswald efficiency factor; landing configuration	e	0,7
Aerodynamic efficiency in take-off configuration	E_{TO}	10,32
Number of engines	n_E	3
Climb gradient	$\sin(\gamma)$	0,027
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,186

Missed approach

Lift coefficient, landing	$C_{L,L}$	2,19
Lift-independent drag coefficient, clean	$C_{D,0}$ (Missed approach)	0,020
Lift-independent drag coefficient, flaps	$\Delta C_{D,flap}$	0,054
Lift-independent drag coefficient, slats	$\Delta C_{D,slat}$	0,000
Choose: Certification basis	JAR-25 resp. CS-25	**no**
	FAR Part 25	**yes**
Lift-independent drag coefficient, landing gear	$\Delta C_{D,gear}$	0,015
Profile drag coefficient	$C_{D,P}$	0,089
Aerodynamic efficiency in landing configuration	E_L	7,01
Climb gradient	$\sin(\gamma)$	0,024
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,221

2) Maximum Aerodynamic Efficiency

Constant parameters

Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Oswald eff. factor, clean	e	0,85	

Specifications

Mach number, cruise	M_{CR}	0,8	
Aspect ratio	A	9,78	
Bypass ratio	μ	4,50	
Wing loading	m_{MTO}/S_W	468	kg/m²
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,276	

Variables

	V/V_{md}	**1,2**

Calculations

Zero-lift drag coefficient	$C_{D,0}$	0,019
Lift coefficient at E_{max}	$C_{L,md}$	0,71
Ratio, lift coefficient	$C_L/C_{L,md}$	0,683
Lift coefficient, cruise	C_L	0,485
Actual aerodynamic efficiency, cruise	E	17,11
Max. aerodynamic efficiency, cruise	E_{max}	**18,37**

Newton-Raphson for the maximum aerodynamic efficiency

Iterations	1	2	3
f(x)	0,25	-0,01	0,00
f'(x)	-0,10	-0,11	-0,11
E_{max}	16	18,48	18,37

3) Specific Fuel Consumption

Constant parameters			
Ratio of specific heats, air	γ	1,4	
Earth acceleration	g	9,81	m/s²
Air pressure, ISA, standard	p_0	101325	Pa
Fuel density	ρ_{fuel}	800	kg/m³

Specifications			
Range	R	6450	NM
Mach number, cruise	M_{CR}	0,8	
Bypass ratio	μ	4,50	
Thrust-to-weight ratio	$T_{TO}/(m_{MTO}*g)$	0,276	
Available fuel volume	$V_{fuel,available}$	23,1625	m³
Maximum take-off mass	m_{MTO}	33113	kg
Mass ratio, landing - take-off	m_{PL}/m_{MTO}	0,054	
Mass ratio, operating empty - take-off	m_{OE}/m_{MTO}	0,498	

Calculated values			
Actual aerodynamic efficiency, cruise	E	17,11	
Cruise altitude	h_{CR}	11500	m
Cruise speed	V_{CR}	236	m/s

Mission fuel fraction			
Type of aeroplane (according to Roskam)	**Business jet**		
Fuel-Fraction, engine start	$M_{ff,engine}$	**1,000**	
Fuel-Fraction, taxi	$M_{ff,taxi}$	**0,997**	
Fuel-Fraction, take-off	$M_{ff,TO}$	**0,994**	
Fuel-Fraction, climb	$M_{ff,CLB}$	**0,994**	
Fuel-Fraction, descent	$M_{ff,DES}$	**0,994**	
Fuel-Fraction, landing	$M_{ff,L}$	**0,994**	

Calculations			
Mission fuel fraction (acc. to PL and OE)	m_F/m_{MTO}	0,448	
Mission fuel fraction (acc. to PL and OE)	M_{ff}	0,552	
Distance to alternate	$s_{to\_alternate}$	100	NM
Distance to alternate	$s_{to\_alternate}$	185200	m
Choose: FAR Part121-Reserves	domestic	**yes**	
	international	**no**	
Extra-fuel for long range		5%	
Extra flight distance	s_{res}	185200	m
Loiter time	t_{loiter}	2700	s
Specific fuel consumption	SFC	**1,80E-05**	kg/N/s

4) Verification Specifications

Maximum lift coefficients

General wing specifications	*Airfoil type:*	**NACA 4 digit**	
Wing span	b_W	26,29	m
Structural wing span	$b_{W,struct}$	29,92	m
Wing area	S_W	70,7	m²
Aspect ratio	A	9,78	
Sweep	φ_{25}	28,5	°
Mean aerodynamic chord	c_{MAC}	3,8	m
Position of maximum camber	$x_{(y\_c),max}$	25	%c
Camber	$(y_c)_{max}/c$	4	%c
Position of maximum thickness	$x_{t,max}$	30	%c
Relative thickness	t/c	11,3	%
Taper	λ	0,22	
General aircraft specifications			
Temperature above ISA (288,15K)	ΔT_L	0	K
Relative density	σ	1	
Temperature, landing	T_L	273,15	K
Density, air, landing	ρ	1,225	kg/m³
Dynamic viscosity, air	μ	1,72E-05	kg/m/s
Speed of sound, landing	a_{APP}	331	m/s
Approach speed	V_{APP}	54,53	m/s
Mach number, landing	M_{APP}	0,16	
Mach number, cruise	M_{CR}	0,8	
Calculations maximum clean lift coefficient			
Leading edge sharpness parameter	Δy	2,9	%c
Leading edge sweep	φ_{LE}	32,2	°
Reynoldsnumber	Re	1,5E+07	
Maximum lift coefficient, base	$c_{L,max,base}$	1,56	
Correction term, camber	$\Delta_1 c_{L,max}$	0,19	
Correction term, thickness	$\Delta_2 c_{L,max}$	0,00	
Correction term, Reynolds' number	$\Delta_3 c_{L,max}$	0,035	
Maximum lift coefficient, airfoil	$c_{L,max,clean}$	1,787	
Lift coefficient ratio	$C_{L,max}/c_{L,max}$	0,78	
Correction term, Mach number	$\Delta C_{L,max}$	0,00	
Lift coefficient, wing	$C_{L,max}$	1,40	

Calculations increase of lift coefficient due to flaps		**1 flap type**	
Correction factor, sweep	K_φ	0,85	
• Flap group A			
Double-slotted flap	$\Delta c_{L,max,fA}$	1,37	
Use flapped span	b_W,fA	**14,85**	m
Percentage of flaps allong the wing		50%	
Increase in maximum lift coefficient, flap group A	$\Delta C_{L,max,fA}$	0,58	
Increase in maximum lift coefficient, flap	$\Delta C_{L,max,f}$	0,58	

Calculations increase of lift coefficient due to slats		**1 slat type**	
Sweep angle of the hinge line	$\varphi_{H.L.}$	**64**	°
• Slat group A			
Handley Page slat	$\Delta c_{L,max,sA}$	0,59	
Use slatted span	b_W,sA	**25,55**	m
Percentage of slats allong the wing		85%	
Increase in maximum lift coefficient, slat group A	$\Delta C_{L,max,sA}$	0,22	
Increase in maximum lift coefficient, slat	$\Delta C_{L,max,s}$	0,22	

Wing		
Verification value maximum lift coefficient, landing	$C_{L,max,L}$	**2,17**
RE value maximum lift coefficient, landing		3,70
Verification value maximum lift coefficient, take-off	$C_{L,max,TO}$	**1,27**
RE value maximum lift coefficient, take-off		2,17

41%

Aerodynamic efficiency

Real aircraft average	k_{WL}	2,83	
End plate	$k_{e,WL}$	1,06	
Span	b_W	26,29	m
Winglet height	h	**1,05**	m
Aspect ratio	A	9,78	
Effective aspect ratio	A_{eff}	10,34	
Efficiency factor, short range	k_E	17,25	
Relative wetted area	S_{wet}/S_W	**6,20**	
Verification value maximum aerodynamic efficiency	E_{max}	**22,3**	
RE value maximum aerodynamic efficiency		18,37	

21%

Specific fuel consumption (Herrmann 2010)

Cruise Mach number	M_{CR}	0,800	
Cruise altitude	h_{CR}	11500	m
By Pass Ratio	μ	4,50	
Take-off Thrust (one engine)	$T_{TO,one\ engine}$	29,90	kN
Overall Pressure ratio	OAPR	**15,88**	
Turbine entry temperature	TET	**1252,44**	
Inlet pressure loss	$\Delta P/P$	2%	
Inlet efficiency	η_{inlet}	0,95	
Ventilator efficiency	$\eta_{ventilator}$	0,74	
Compressor efficiency	$\eta_{compresor}$	0,81	
Turbine efficiency	$\eta_{turbine}$	0,82	
Nozzle efficiency	η_{nozzle}	0,94	
Temperature at SL	T_0	288,15	K
Temperature lapse rate in troposhpere	L	0,0065	K/m
Temperature (ISA) at tropopause	T_S	216,65	K
Temperature at cruise altitude	T(H)	216,65	K
Dimensionless turbine entry temperature	ϕ	5,78	
Ratio of specific heats, air	γ	1,40	
Ratio between stagnation point temperature and temperature	υ	1,13	
Temperature function	χ	1,36	
Gas generator efficiency	η_{gasgen}	0,98	
Gas generator function	G	1,51	
Verification value specific fuel consumption	SFC	0,94	kg/daN/h
Verification value specific fuel consumption	SFC	**2,61E-05**	kg/N/s
RE value specific fuel consumption	SFC	1,80E-05	kg/N/s

45%

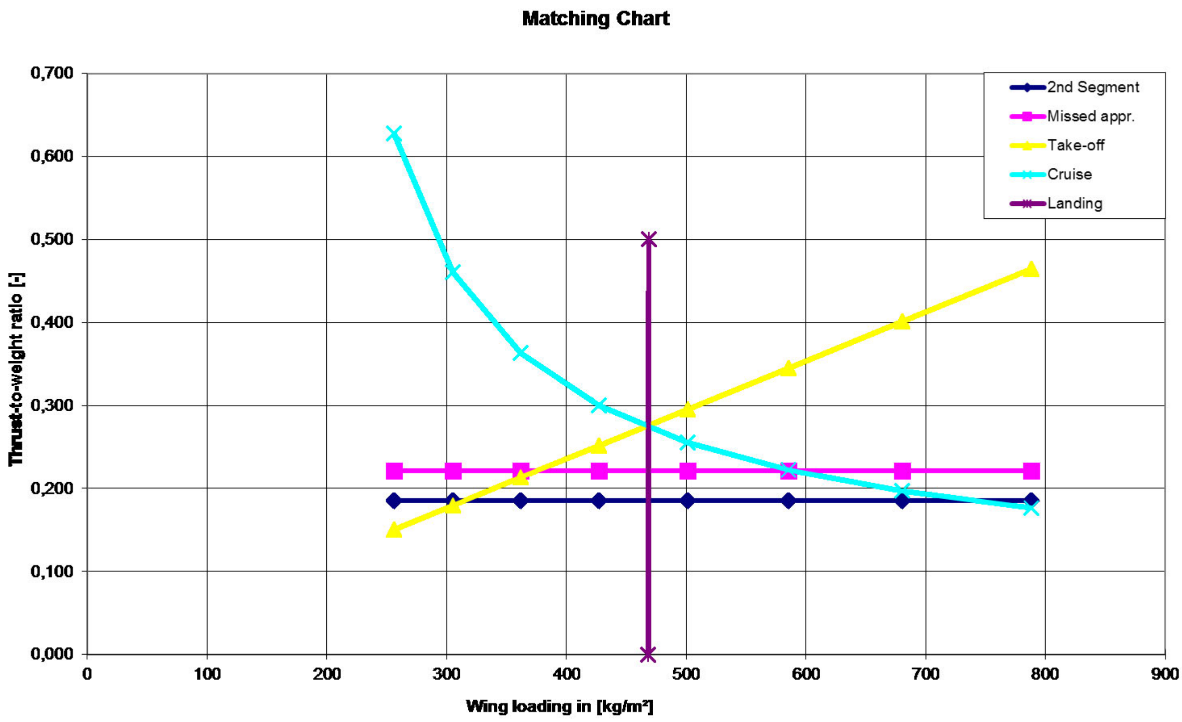

Figure I.1 Matching chart – Falcon 8X