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Methodology for the Design of Leading Edge Devices Applied to Variable Camber

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Methodology for the Design of Leading Edge Devices Applied to Variable Camber

Supervisor: Professor J.P. Fielding

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i. Abstract

This thesis will describe a new and innovative way of approaching the design of leading edge devices; this done using new tools and state of the art software. This innovative design methodology for leading edge devices has a special focus on the application of variable camber technology.
A precise description is given of the way to check that methodology acts as a means of validation.

A case study shows how to apply this methodology and examines a variable camber application. This case study provides the basis for the understanding of how to apply the design methodology and give results generated from its different parts. An analysis of the structure and aerodynamic performances of a new type of leading edge device was performed in addition to moving the overall design towards an optimum solution in term of mass, reliability and cost.

The case study is showing that using a variable camber leading edge device could be beneficial as it proves a better option than more classical concepts. Some of the results show that the aerodynamics implications of using this type of device at the leading edge can be beneficial in cruise and also during take off and landing. Also on the structural side of things, it is possible to see that the deployment trajectory as been optimised to fit to the required trajectory and the structure is able to resist critical loading.

Finally there is a discussion on the obtained results and on the overall methodology to make conclusions on the overall meaning of this research and the possible impact of the new design methodology as well as implication on the design of variable camber leading edge devices.

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vi. Symbols

Symbol	Description	Unit
A	Area	[m^2]
AR	Aspect Ratio	[--]
b	Wing span	[m]
C	Speed of sound	[m.s-1]
c	Wing chord	[m]
C_{Assy} =	Overall cost of one type of assembly	[£]
$c_{average}$	Average panel chord	[m]
C_{comp} =	Overall cost of one type of component	[£]
C_{D0}	Drag coefficient at zero lift	[--]
$C_{L\ appr}$	Lift coefficient during approach	[--]
C_L	Lift coefficient	[--]
C_{Lmax}	Maximum lift coefficient for a given flap configuration	[m/s]
C_{Manuf} =	Cost of manufacturing 1 type of component	[£]
C_{MAT} =	Cost of material to buy to manufacture 1 type of component	[£]
C_P	Pressure coefficient	[--]
Cr	Root chord	[m]
C_{SPEC} =	Specific cost to weight coefficient	[£/Kg]
Ct	Tip chord	[m]
d	Type of device designator	[--]
D_{Comp} =	Dimension of components	[m]
$F_{central}$	Total failure rate for central wing section	[per 10^6h]
F_{COMP}	Component failure rate	[per 10^6h]
$F_{COMP,cycle}$	Specific component failure rate	[per 10^6 cycle]
$F_{COMP,time}$	Specific component failure rate	[per 10^6h]
F_D	Failure rate for one type of device	[per 10^6h]
F_{Dd}	Specific device type failure rate	[per 10^6h]
F_{inb}	Total failure rate for inboard wing section	[per 10^6h]
F_{outb}	Total failure rate for outboard wing section	[per 10^6h]

F_P	Panel failure rate	[per 10^6h]
F_{Pd}	Specific panel failure rate	[per 10^6h]
F_s	Total failure rate for one wing section	[per 10^6h]
F_{total}	Total failure rate for the whole aircraft LE system	[per 10^6h]
F_{wing}	Total failure rate for the LE system of one wing	[per 10^6h]
K	Adiabatic index also called γ	[--]
L	Total lift force	[N]
M	Mach number	[--]
M_A	Specific mass ratio per unit area	[kg/m^2]
MAC	Mean aerodynamic chord	[m]
M_C	Specific mass ratio per chord unit	[kg/m]
M_{COMP}	Mass of one type of component per panel	[kg]
$M_{COMP,Specific}$	Specific mass of component	[kg]
M_{Dd}	Mass of one device	[kg]
M_{Di}	Mass of each types of device	[kg]
$M_{LE\ Central}$	Mass of the LE system for the central section	[kg]
$M_{LE\ inboard}$	Mass of the LE system for the inboard section	[kg]
$M_{LE\ outboard}$	Mass of the LE system for the outboard section	[kg]
$M_{LE\ Section}$	Mass of the LE devices of one section (in/outboard)	[kg]
$M_{LE\ total}$	Mass of the LE system for the aircraft	[kg]
$M_{LE\ W}$	Mass of the LE system for one wing	[kg]
M_P	Mass of one panel	[kg]
M_{Pd}	Specific panel mass for each type of device	[kg]
MTOW	Maximum take-off weight	[kg] or [N]
N_{Pd}	Number of panel for each device/section	[--]
N_{Pd}	Number of panel of each type of device	[kg]
P	Pressure	[F/m^2]
PA	Projected panel area	[m^2]
R	Universal gas constant	[J/kg.K]
R/C	Climb rate	[--]
$R_{B"F}$ =	"Buy to Fly" ratio	[--]

Re	Reynolds Number	[--]
R_{Manuf} =	Rate for manufacturing operations	[£/m]
s	Semi-span	[m]
S	Wing area	[m^2]
T	Temperature	[K]
t	Wing (profile) thickness	[m]
t/c	Airfoil thickness ratio	[--]
V	Free stream velocity	[m/s]
V_2	Climb speed	[m/s]
V_{LOF}	Lift-off speed	[m/s]
V_{MC}	Minimum control speed	[m/s]
V_{MU}	Minimum unstuck speed	[m/s]
V_{S1g}	Stall speed in steady flight	[m/s]
V_{Smin}	Minimum dynamic stall speed in steady flight	[m/s]
W_{LE}	Weight of leading edge	[kg] or [N]
$\bar{c}$	Average chord	[m]

Greek symbols:

α	Quantity of component / Number of component	
γ	Adiabatic Index (gamma)	[--]
γ	Ratio of specific heat (generally 1.4 for perfect gas)	[--]
ϑ	Temperature (theta)	[C°]
Λ	Sweep angle (lambda)	[°]
λ	Taper ratio (lambda)	[--]
μ	Coefficient of viscosity	[kg/s.m]
ρ	Density of air (rho)	[kg/m3]

vii. Abbreviations

AIAA	-American Institute of Aeronautics and Astronautics
Al	-Aluminium
AR	-Aspect Ratio
ATRA	-Advanced Transport Regional Aircraft
ATVCW	-Advanced Technology Variable Camber Wing
CAE	-Computer Aided Engineering
CFD	-Computational Fluid Dynamics
CG	- Centre of Gravity
DOC	-Direct Operation Cost
ESDU	-Engineering Sciences Data Unit
FAA	-Federal Aviation Administration
FAR	-Federal Airworthiness Regulation
FEA	-Finite Element Analysis
HLFC	-Hybrid Laminar Flow Control
L/D	-Lift to Drag ratio
LE	-Leading Edge
MAW	-Mission Adaptive Wing
Mg	-Magnesium
MTOW	-Maximum Take-Off Weight
NASA	-National Aeronautics & Space Administration
RAEVAM	-Royal Aircraft Establishment Variable Camber Mechanism
SAWE	-Society of Allied Weight Engineers
STOL	-Short Take-Off and Landing
TE	-Trailing Edge
Ti	-Titanium
VC	-Variable Camber
VCF	-Variable Camber Flap
VCW	-Variable Camber Wing

Chapter 1

1. Introduction & Objectives

1.1. General

At the present time, airline companies have to deal with the soaring price of oil whilst operating within a very competitive market; therefore they are always looking for improvements from the aircraft manufacturers (Boeing, Airbus...). They want to reduce their Direct Operation Costs (DOC) as well as improve the operational flexibility of their fleet. There are now more technologies and greater computational power available to the aircraft manufacturers' design office. However, there have only been limited changes to wing geometry. Most of the current research is concentrated on limiting the noise and pollution but not so much research is based on the design for an optimally cambered wing, and especially the Variable Camber Wing (VCW). There is thus space for investigation into the design and the efficiency of VCW [1], but due to the limitations of classical materials and the low level of technologies available in the past nobody has managed to create a practical design for a potential VCW. With today's tools and software it is much more likely that there will be a breakthrough in the design of Variable Camber (VC) devices which will radically alter the way we design wings. With this we will be able to control, and manage the position of such devices at the different flight conditions (cruise, take off and landing). Using VC will show improvements in the performance of the aircraft, as the pilots will be able to use the VC device to reduce the drag during the flight.
It has been found that much work has been done in the past regarding the study of high-lift devices but less attention has been given to VCW designs

1.2. Research background

The Cranfield College of Aeronautics has been carrying out research on such wings for more than 20 years and has produced some results described in different PhD theses.

The current research builds on work done by other people, but none of them studied the effect and design of LE device applied to VC in particular. Ammoo[2] developed a design methodology applied to variable camber flaps but did not study the effect of the LE. Precedent work by Macci [3] was done on the structural and mechanical aspect of VCW. There was also some fluid dynamic work on VCW done by Edi[4] (effect of hybrid laminar flow and variable camber) and by Mc Kinnon[5], but in all the research done at Cranfield University or by others [1, 6], there always have been questions and recommendations for further research on the effects and the design of LE devices.

1.3. Research objective

The main objective of this research work is to develop a generic methodology for the design of LE devices, including application to Variable Camber for passenger or transport subsonic aircraft. This includes reviewing current design for these applications and developing new solutions. Ideally there will be a new design methodology created using the results of both the fluid dynamic and structural analysis. It is expected that this will be validated and compared to research done by other experts on this field.

1.4. Research method

- Review of existing mechanisms and methodologies

The research starts with an extensive review of existing leading edge VC design and design of other leading edge devices. This includes the different design schemes and the different technologies available. There is also a short description of the effect on the aerodynamic performance of the aircraft, for each leading edge device type of design. This will be followed by a detail review of current design methodologies.

- Development of a design methodology

A new design methodology is described; this methodology is for the design of LE devices using a possible VC concept. It covers the early design stages and incorporates aerodynamic analysis, mechanism design and structural analysis. For this methodology, new specific tools have been developed and used to shorten the time to design. The methodology process and the different tools used are described in more detail in this chapter.

- Validation of the methodology

The methodology incorporates different steps which cover different aspects, such as mechanism design, fluid dynamics or structural analysis. Each one of them has to be validated against experimental results or against other theoretical results obtained by experts in these fields of research. This validation process will show the quality of the different processes and tools used for the methodology.

- Case study

A case study is performed to test the methodology, and to show how it is possible to apply the methodology to real case. The case study chosen was a regional transport aircraft.

- Discussion

This chapter analyses and discusses the results obtained in the case study. There will be extensive and detailed comments on the results obtained, but also on the design process linked with the methodology.

- Conclusions and recommendations

A conclusion is made on the progress of the study, and recommendations for further work have been included to improve the methodology and to extend the testing of new LE devices.

Chapter 2

2. Technical Review

2.1. Introduction

The following pages describe the state of the art research and development for the design of LE devices. The review of these different designs is separated into two parts. The first part explains the effect and the theory behind each concept, and the second part focuses on the technical description of the mechanical solutions with a detailed explanation of each concept and the way they are designed. The following paragraph will described the different concepts from the simplest one to the more complex ones.

2.2. Theory background

2.2.1 Effect of leading edge devices

LE devices allow the pilot to influence the change of shape and geometry of the leading edge of the wing. By doing so, the pilot can change the aerodynamic performance of the aircraft. Each concept has different advantages and drawbacks, and each concept will be described in the following paragraphs.

2.2.2 Fixed slot

Fixed slot are no more than a slat at a fixed position (see Figure 1) with a fixed gap between the main airfoil and the leading edge. The reason behind this design is to delay the stall (by increasing the angle of attack before stall) compared to a classical airfoil without a slot (see Figure 2).

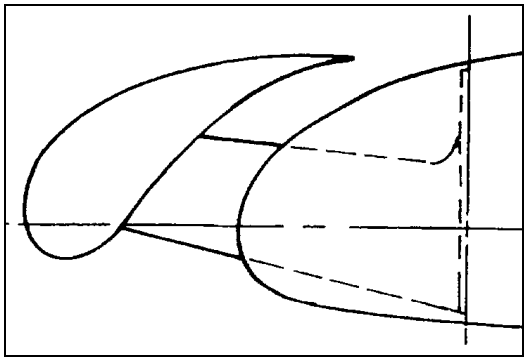

Figure 1 - Fixed slot [7]

Stalling is caused by the breakdown of the steady streamline flow. On slotted wing the air flows through the gap in such a way to keep the airflow smooth, following the surface of the airfoil and continuing to provide lift until a greater angle is reached [8].
Historically this concept was the first one developed to improve the airfoil aerodynamic performances using the LE. Handley Page was the first one to use and implement such a concept in 1919.This concept has since been successfully and regularly used on Short Take-Off and Landing (STOL) aircraft with low speed**.**

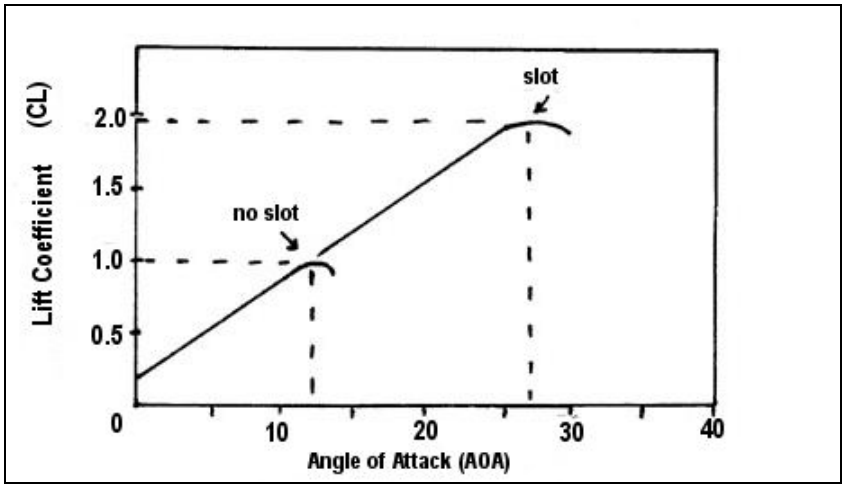

Figure 2 - Lift with and without slot [9]

One of the drawbacks of the slot is the drag penalty occurring during cruise condition. As the slot is still present, the airflow is perturbed and so it generates drag which is unacceptable for high performance subsonic aircraft[10]. A Possible improvement would be to have moving slats, as discussed in the next paragraph.

2.2.3 Moving slat

Moving slats devices are of two kinds:

- Controlled= moves backward and forward by a control mechanism
- Automatic= moves on their own due to the air pressure (suction near the LE)

Slats are small but highly cambered airfoils forward of the wing leading edge which experience large suction forces per unit of area[7]. This phenomenon is used for the deployment of the automatic slat as the suction pulls the slat out of its initial position.
Controlled slats are deployed using a control mechanism (slat-track mechanism generally) in order to create the ideal slot to improve the aerodynamic performance of the aircraft.

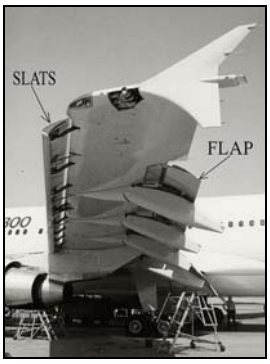

Figure 3 - A300 slats and flap[11]

Slotted LE devices have high maximum lift capabilities and as a result they produce higher drag, so they are best used for landing. The controlled slats are normally deployed to one position to increase lift during landing, however some modern commercial aircraft use a 3-position slat deployment system to have an intermediate position. This

intermediate position is a sealed configuration (small deflection) to produce lift but less drag than the slotted configuration, and so they are used during the take –off phase[12]. Moving slats can be of two sorts, a two or a three position slat, the latter using the extra slat position for the take-off configuration[10].

The opening of the slot can be delayed or hastened by "vents" at the trailing edge or leading edge of the slat respectively and there may be some kind of spring or tensioning device to prevent juddering, which may be otherwise likely to occur[8].

2.2.4 Droop nose

Droop noses are less effective than slats but can have advantages for specific applications. The simple droop nose design generates an abrupt change of chord direction and so the air flow has difficulty turning the corner without separating[7]. Therefore it creates a turbulence and separation of the airflow, and a loss of efficiency in terms of lift and an increased drag. This concept is mainly used for fighter aircraft as they fly at supersonic speed (M>1). The droop nose become efficient as the air flow characteristic is changing (compare to subsonic), and a much higher LE sweep angle generates a stable vortex on the upper surface of the wing, which provides lift [10].

2.2.5 Krueger flap

A Krueger flap is a panel on the lower side of the LE which rotates down and forward which generally helps to generate high-lift performance (see Figure 6 in page 10). This kind of flap is used for landing, and is retracted during cruise. It generates extra lift when deployed, and it is generally used at slow speed (approach) for large aircraft to provide some stall protection. Krueger flaps can be designed to be sealed or vented when deployed; the vented option is used to delay the stall. During normal operation there is generally a stagnation bubble on the upper aft portion of the Krueger flap[10], and the flow is attached only over a small range of angle of attack. Krueger flaps are sometimes designed using VC technology, but this increases the total weight (as explained later in this chapter). It is also possible to add a folding bull nose to extend the flow attachment over a larger angle of attack.

2.2.6 Variable camber

Variable camber concepts are used to improve the efficiency of the airfoil compared to the improvements provided by the other concepts described above. With VC the air flow is not separated, since the airfoil top and bottom surface is kept smooth and hence there is no abrupt change of chord direction. All the changes to the airfoil profile are kept smooth to avoid separation of the air flow. The effect of camber is to increase the minimum value of C_{D0} slightly but to increase the value of C_L. For the same value of angle of attack a VCW would have a much higher lift to drag ratio (L/D)[13]. This type of concept can be used at both the LE and TE of the wing.

2.3. Different kinds of leading edge mechanism designs

There are many different types of LE devices, including slats to flaps or Krueger flaps. It is possible to create infinity of concepts, but it is always important to design a system which has a low weight and is easy to maintain. The following paragraphs outline a few examples of LE device designs which lead to VC applications.

2.3.1 Slat track mechanism

The slat track mechanism is one of the most common mechanisms utilised for the design of LE devices. This mechanism has a track which guides the deployment of the slat and so the deployment trajectory is the exact replica of the track path (Figure 4). Slat-track mechanisms generally have three settings on modern aircraft, as previously explained: stowed, takeoff and landing. The tracks are constructed to optimize the configuration for each manoeuvre, so the slat is sealed for take-off, and vented for landing

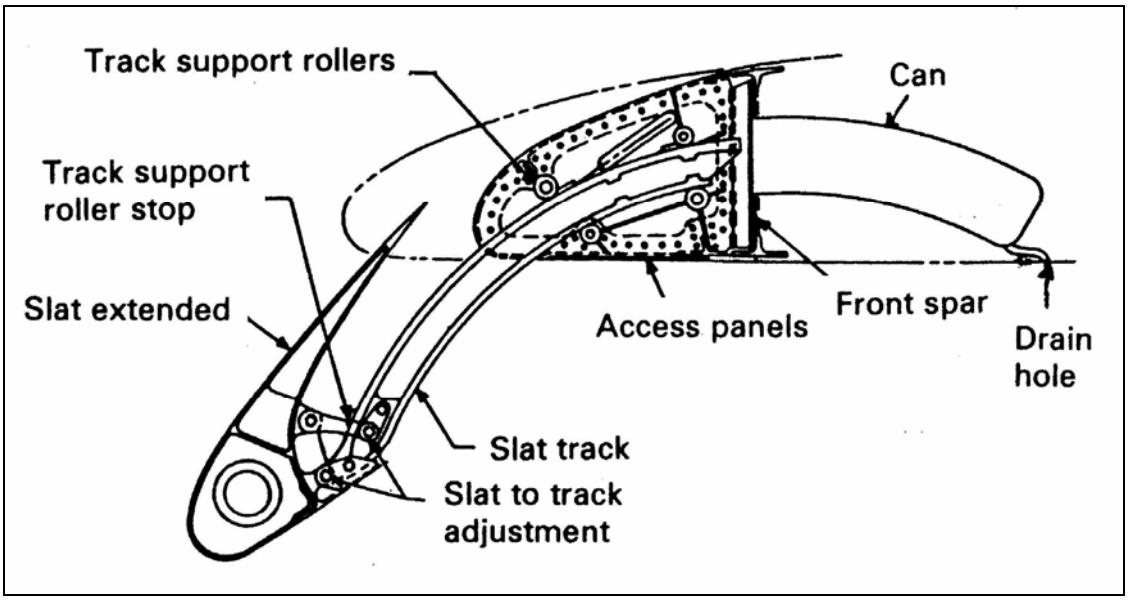

Figure 4 - Slat track mechanism[7]

This system is often used because it does not require much development and research. The principle behind this type of mechanism is very simple: an actuator pushes the slat out of its initial position to follow the track path. The drawbacks of this system include the amount of rollers necessary to keep the slat in the exact position while being deployed, as well as the volume (or space) occupied by the track can (Figure 4). In this case there is a loss of space for the fuel tank located within the wing, and so a decrease in the length of the flight. Also this concept requires several parts including bearings, which need to be replaced regularly, and so decrease the overall reliability of the system.

There are two major types of slats track designs in the market today. One is a constant-chord slat (Boeing 737-100), the second one is slightly tapered (Boeing 737-300 & Airbus A310 A320 A330).

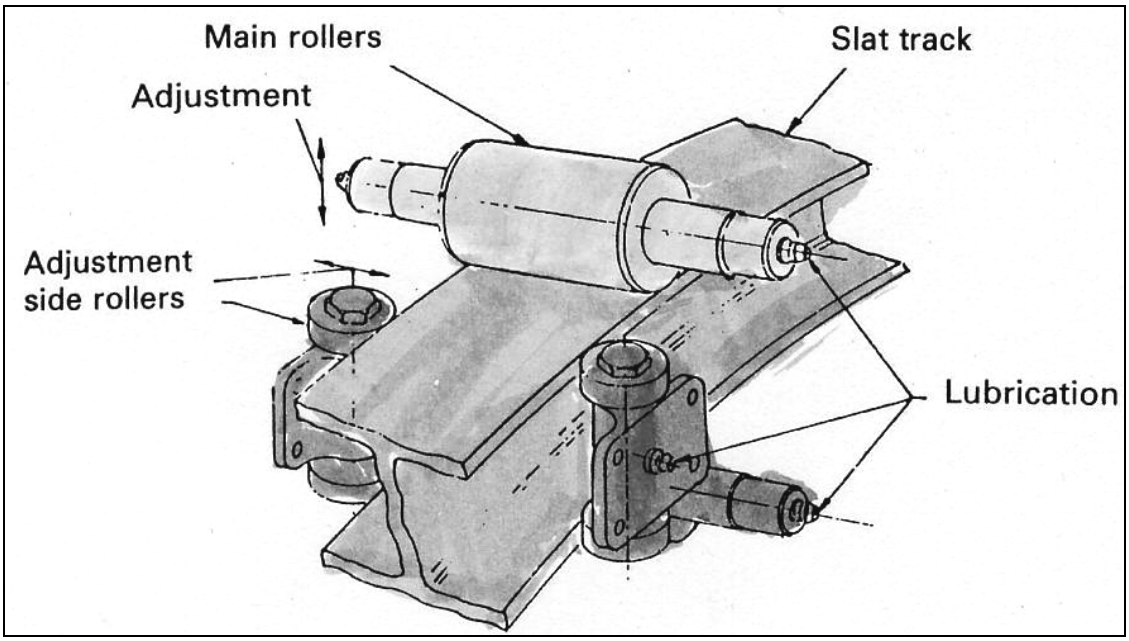

Figure 5 - Roller arrangement [7]

Another feature of slat design is that the motion of deployment (in 3D) can be either cylindrical or conical. The cylindrical motion is the best for constant (or nearly constant) chord slats, and the conical motion the best for tapered slats. Cylindrical slat motion only requires identical tracks associated with a simple actuation system, whereas the conical motion requires different track radii and a more complex actuation system. It is clear that the cylindrical motion will greatly reduce the cost and complexity of the design, but it might not be as efficient in terms of aerodynamics, as the conical motion with tapered slat [10].

The slat deployment could also be reduced, with a smaller LE device angle and so could produce a shift of the C_L (lift coefficient) versus alpha to the left. This is called a shallow slat, but the problem generated with this idea is that a larger gap is necessary[14] and so more research into the mechanical feasibility of this concept would be required to show the full efficiency of this idea.

Due to the high aerodynamic loads applied to the slat, it is required to have closely spaced ribs for the slat to be able to resist different flying conditions. It is also current practice to use a thick skin made of Aluminium-alloy (Al-Alloy) 2024-T3 or similar, with a skin of no less than 1.6mm thickness. This is for wing protection against rain erosion, hail damage, and bird strikes.
The track is normally made of Titanium (Ti) or high strength steel. These materials are used to react against the very high bending loads when the slats are extended [15].

2.3.2 Krueger flap

The Krueger flap is formed by moving the front lower surface, generally an almost flat panel, of the aerofoil outwards and forwards about a hinge point located on the airfoil surface at the leading edge (paragraph 2.2.5), or slightly behind and above it (Figure 6). When extended, the flap forms a forward extension of the upper surface thereby increasing chord as well as camber.

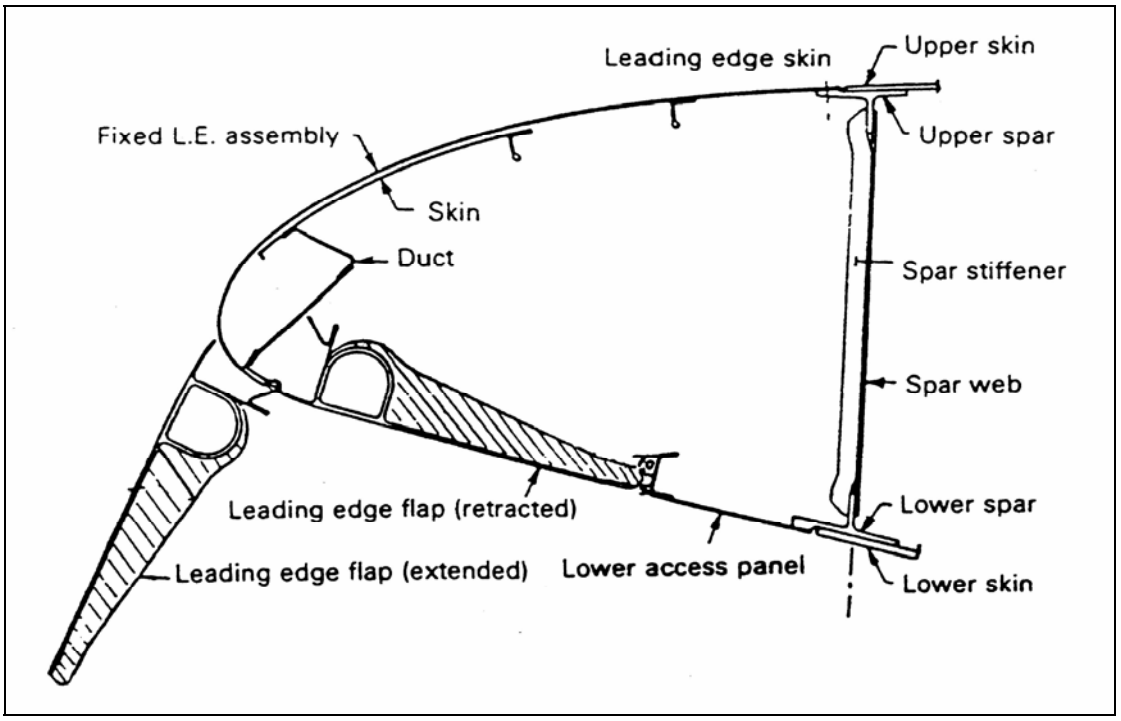

Figure 6 - Simple Krueger flap [7]

There are different ways of designing a Krueger flap but all have the same effects on the aerodynamics. Generally Krueger flaps are used in the inboard part of the wing in combination with outboard slats to obtain a positive longitudinal stability. For example the Boeing 707-320 uses three Krueger flaps inboard and five outboard slats.

There are two existing sorts of Krueger flaps, the rigid Krueger flap as explained below, and the VC Krueger flap, which will be explained in the following pages.

Krueger flaps are normally simpler and lighter structures than slats, and they are also easier to manufacture. They are generally made of one panel of Al alloy, Magnesium (Mg) or composite materials. This design also has the advantage of keeping the fuel tank volume as original compared to track-cans with slats. When fully retracted, the Krueger flaps are fully sealed with the bottom surface of the LE and therefore there is no air leakage [7] (Figure 7). Therefore, this concept does not offer the possibility to have a gap between the flap and the main airfoil.

The advantage of a rigid Krueger flap design is that the flap panels are stiffer than for a VC Krueger flap, which means that less hinges and jointts will be necessary, and so lead to a weight reduction[10], and also an improved reliability.

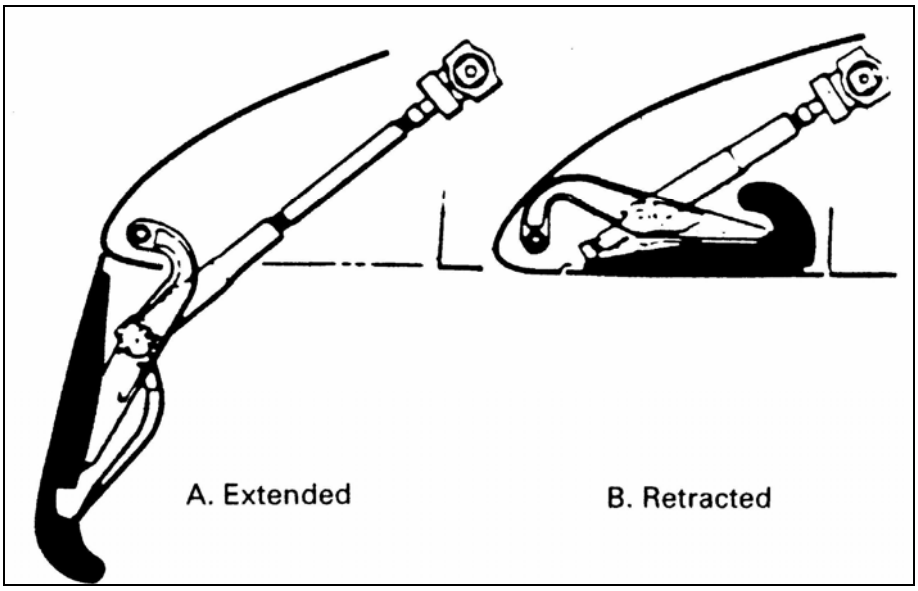

Figure 7 - Krueger flap (folding bull nose) [7]

One improvement for the fixed camber Krueger flap is to use a folding bull nose Krueger flap. This type of flap is used on the Boeing 727 wing with one panel inboard and four outboard slats. It is also used on the Boeing 737 and 747 with one and three inboard flaps respectively.
The folding bull nose is no more than a panel (along the Krueger flap length) which is hinged to the aft end in the stowed position (Figure 8). It is connected with a slave linkage, rotating with the deployment of the Krueger flap [10].

Figure 8 - Picture of bull-nose Krueger flap[16]

The rigid Krueger flap offers some improvement compared to a slat. It is easier to manufacture, it also has simpler actuation system. The only drawback is that it is only used (and useful) during the landing phase. It does not improve the aircraft performance during cruise or take-off as there is no gap between the flap and the main airfoil. When deployed, such a device creates extra drag as well as lift.

2.3.3 Droop nose concept

The droop nose design is generally very simple and rigid, and mostly applied to thin airfoil and supersonic aircraft (ideal for fighter aircraft). However, it could also be applied for transport aircraft with a different kind of design, including a hinge and a rotary actuator (Figure 9). The LE rotates around a hinge and keeps a continuous airfoil profile with a seal at the top and bottom of the profile. This is a very robust and simple design.

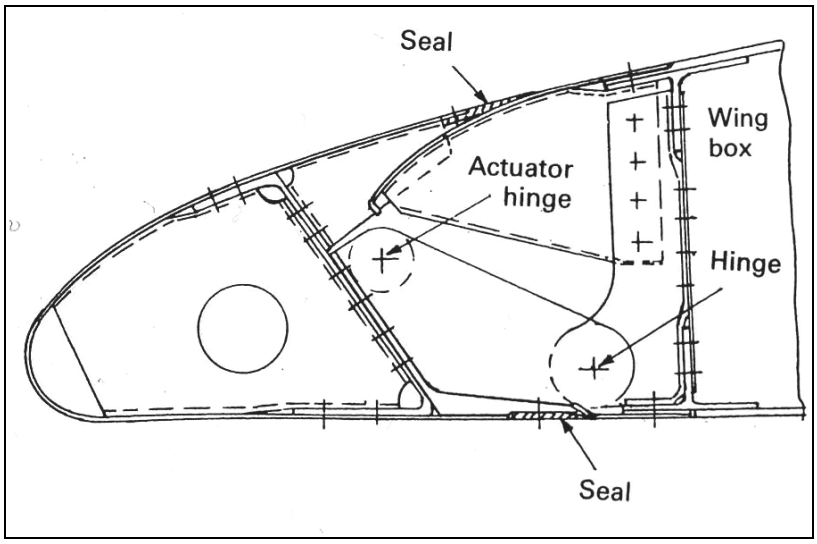

Figure 9 - Droop nose concept for transport aircraft [7]

This concept is generally easy to maintain as it does not include so many parts to make the mechanism.

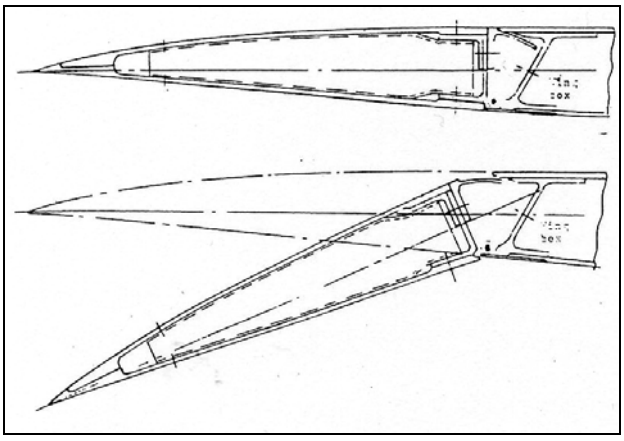

Figure 10 - F-104 fighter LE device [7]

Another example is the F-104 fighter aircraft which uses droop nose technology and started to fly in the 1950's (Figure 10).

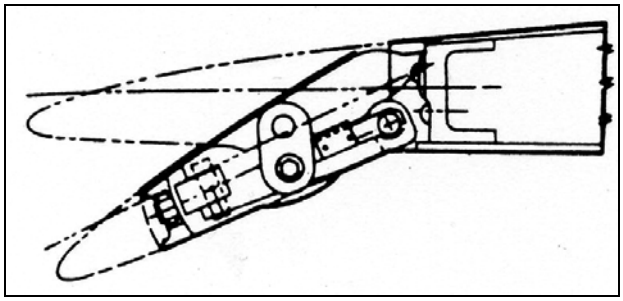

Figure 11 - F-5 fighter LE device[7]

2.3.4 Airbus droop nose concept

In 2004, Airbus developed a totally new concept (Figure 12) for the deployment of their LE droop nose on the A380 and for the A350 [17]. This droop nose is totally sealed at all times during its deployment. As there is no slot the flow does not stay attached to the inboard of the wing for long. Therefore it results in increased lift without increasing the drag, and at the same time helps the wing to stall inboard before stalling outboard[18].
However, compared with other droop nose mechanisms, this one is more complex. It has 4 hinge arms for each panel (with 2 driven and 2 undriven) using rotary actuators to deploy the LE device.
This solution was used on the A380 because of the large depth at the wing root (around 3m at its deepest) as it was not practical to have a slat system due to the weight penalty.

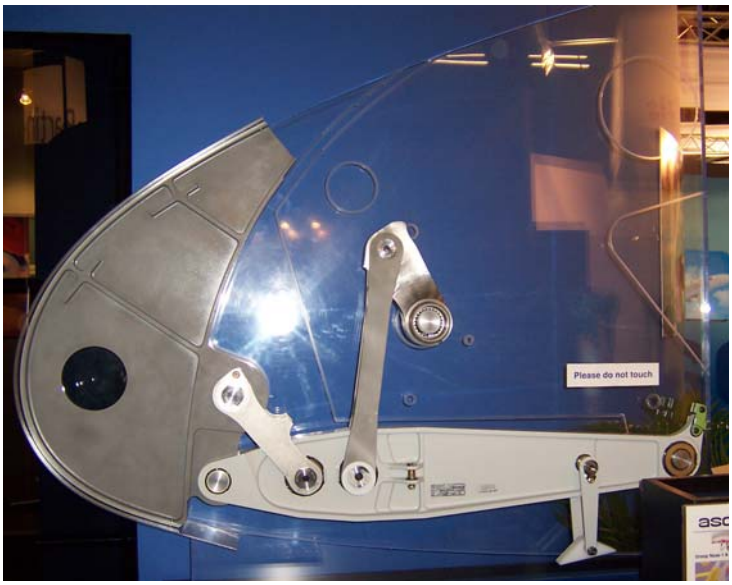

Figure 12 - Airbus A380 droop nose [Lajux]

For this concept Airbus used a material called “Glare” which is made up of alternating layers of aluminium sheets and glass fibre reinforced bond film (thickness between 0.25mm to 0.375 mm)[18]. For this concept Airbus also used Al 7040 for strengthening the structure of the LE, as this alloy is generally stronger than the Al 2024 [19].

2.3.5 Broadbent concept

The Broadbent design is not as simple as the slat track and not as complex as the VC Krueger flaps. This concept is based on a swing arm mechanism (Figure 13) which helps to deploy the slat as well as providing a rotation of this swing arm around an internal axis. This mechanism is being investigated at Cranfield University, with particular emphasis on the kinematics. This concept has been invented and patented by Mr Craig Broadbent[2] (in 2002). He thought of using the swinging motion of an arm to simulate the slat deployment. However this is only a concept at the moment and no aircraft uses this type of mechanism because of some design problems coming from the swinging motion [20]. Former research work has been done by Lo[21] on this concept. He applied it to a fighter aircraft to investigate the aerodynamic efficiency as well as the mechanical feasibility of such an idea.

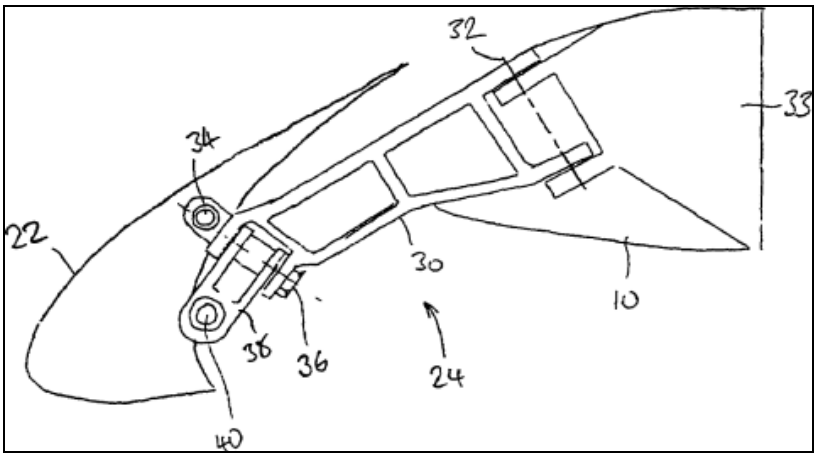

Figure 13 - Broadbent concept (swinging arm mechanism) [20]

2.3.6 Variable camber definition

The aim of so-called “variable camber” or “mission adaptive” wing is to achieve a good high lift or manoeuvrability performance by smooth changes of the airfoil contour, rather than by the more conventional use of movable segments like slats and flaps.

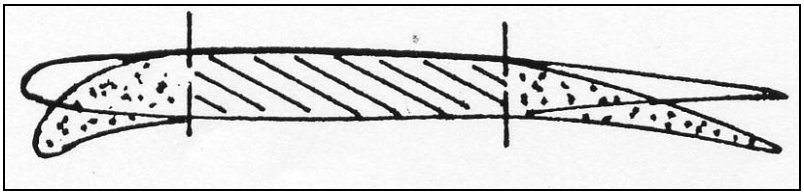

Figure 14 - Variable camber [7]

The variable camber is often achieved by combining a flexible drooped nose with a flexible extending trailing edge flap. The latter may, or may not, incorporate a slot system.

2.3.7 Variable camber LE concept by Boeing

The variable LE concept designed by Boeing in 1980 is mainly to keep a smooth and continuous airfoil surface for lower cruise drag (Figure 15). The same design also meets the requirements of approach high-lift systems. This new design is supposed to continuously optimise the lift to drag ratio and thereby reduce fuel consumption and operating costs.

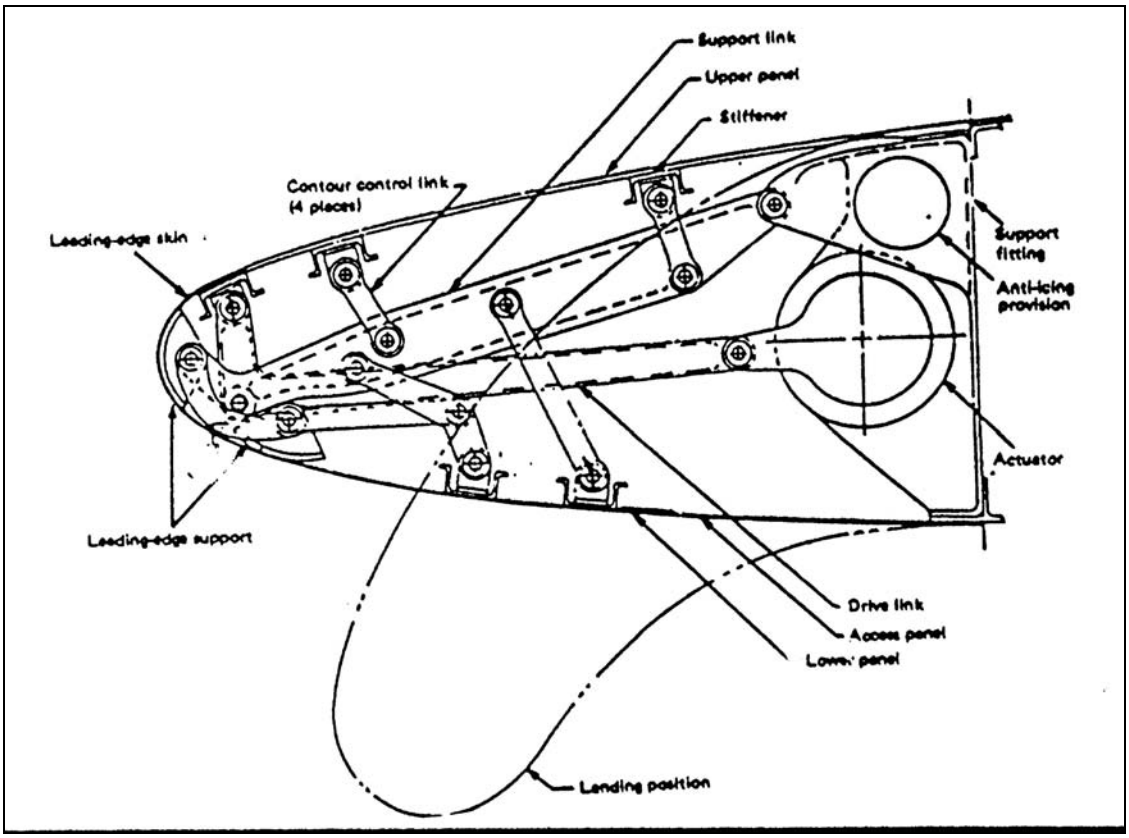

Figure 15 - Boeing VC LE design [3]

This design is extremely efficient as there are no gaps, breaks or overlaps of the skin and the forward skin surface remains the same. The upper and lower surface of the skin is made of fibre glass materials and the leading edge radius is made of stainless steel or titanium for flexibility and erosion resistance[4]. However this design has many components and would prove difficult and expensive to maintain. The impact on the reliability would be too high, and the cost and frequency of maintenance would be a burden for the airlines using such a device on their fleet.

2.3.8 Variable camber Krueger flap by Boeing

This concept was developed around 1958.This system includes the use of a Krueger flap coupled with the variable camber effect by means of a flexible skin (generally made of fibre glass) on the Krueger flap surface. This allows the flap to have a flat surface (cruise condition) and to be able to generate a curved surface when fully deployed (Figure 16). However, all Krueger flaps deploy against the air stream and have a high stowing load at low angles of attack.

The improvement from a Krueger flap to a VC Krueger flap comes with a penalty. The mechanism is now a lot more complex (Figure 17), and the main Krueger panel has to be flexible which requires two stiffeners fixed to the fibreglass panel. Other problems are linked to the rigidity of the Krueger panel span wise. This design requires twice as many panels as a normal Krueger flap, since the fibre glass panels (and stiffeners) tend to distort under the high cruise air load [10].

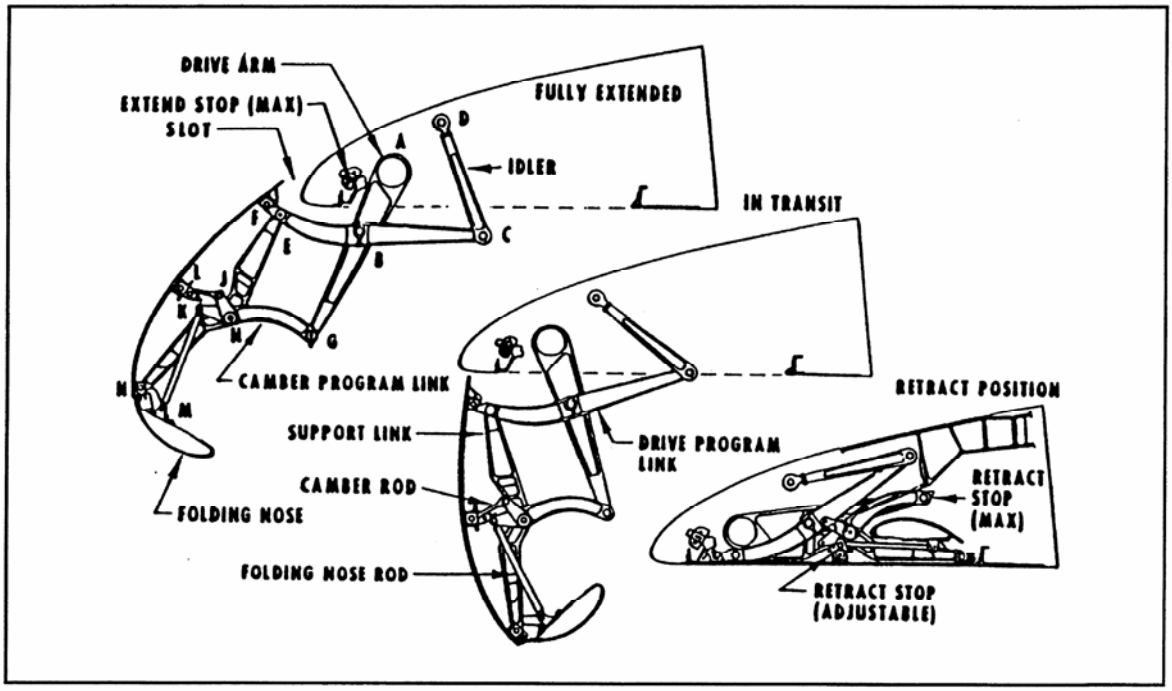

Figure 16 - VC Krueger flap on Boeing 747 [7]

This design is mainly used in commercial aircraft due to the thick wing profile, and also generally used in the inboard part of the wing. For example the Boeing 747-100 has five VC Krueger flaps located between the inboard and outboard engines. This concept does not have the capability of changing camber during cruise condition; this is a high-lift device, which is only used as a two position device (landing and cruise). Therefore the takeoff lift to drag ratio (L/D) is poor and other attempts to make this design a three position device have not been successful [10].

This concept is used on modern aircraft, generally on the larger ones to provide the maximum lift and some drag in order to land on the shortest distance possible.

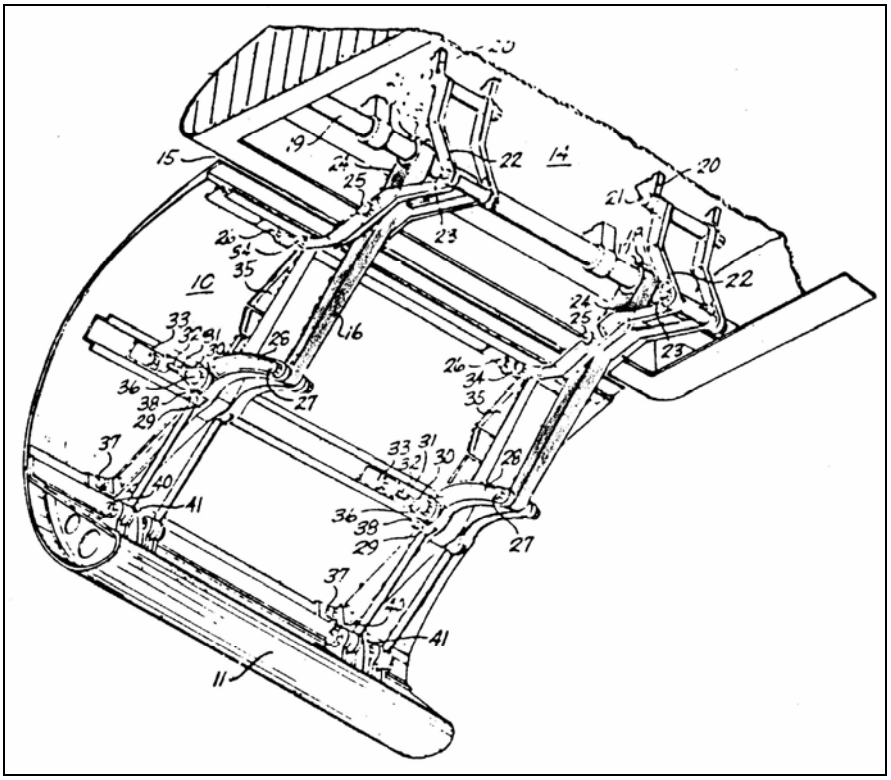

Figure 17 - VC Krueger flap [7]

2.3.9 Mission adaptive wing (MAW)

A "Mission Adaptive Wing" (MAW) has the ability to modify the airfoil camber, span wise camber distribution and also wing sweep in flight. Changes in camber occur from leading to trailing edge and from the wing tip to the root without any wing surface discontinuity. This ability to adapt to different flight conditions means that aircraft using this kind of technology can fly at near-optimum wing configurations at all times (Figure 18). However in the remainder of this research the effect of wing sweep change will not be studied. This is because this research focuses only on the VC type of change and does not incorporate any sweep changes.

In this test bed aircraft the wing leading edge and trailing edges are made of fibreglass skins which were specifically developed to enable it to bend with the motion of the internal structure (Figure 19). Both leading and trailing edge deflections are driven by an internal rotary actuator [2].
This concept has been tested by the National Aeronautics & Space Administration (NASA) on an F-111 experimental aircraft in 1986 (see Figure 18). Rudolph[10] showed that the low speed high-lift characteristics were not so good, and so it is not used on subsonic commercial airliners. However, he also assumed that this concept might find an application as a MAW for subsonic cruise condition over land.

Figure 18 - F-111 (test bed aircraft for MAW) [22]

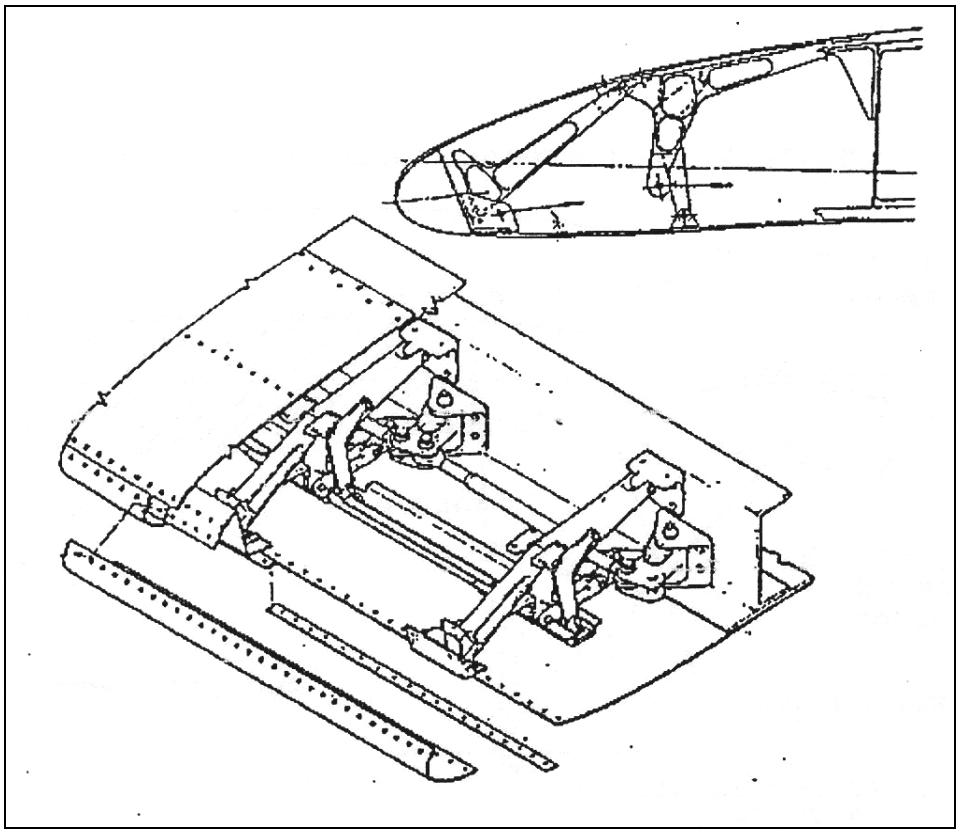

Figure 19 - MAW LE mechanism design [4]

2.3.10 Cranfield variable camber concept

The Cranfield Variable Camber concept was conceived by Spillman [23]. This is a novel method of camber variation by means of rotation and translation of the LE and TE elements (Figure 20). The top surface is kept smooth and continuous to generate a family of cambered airfoil sections[4], which allows airfoil performance to be at their optimum level. A wind tunnel model was developed which included a fixed part (main airfoil) and then the LE and TE as separate modules.

These separate modules had different deflections to be added onto the main airfoil. This model was used in the wind tunnel to mainly study the effect of the trailing edge device. It would appear that the neglect of LE camber deployment during wind tunnel testing on this model results in high LE suction peaks. This could be tolerated at low speed but might generate wave drag at transonic speed which would be a severe penalty [24].

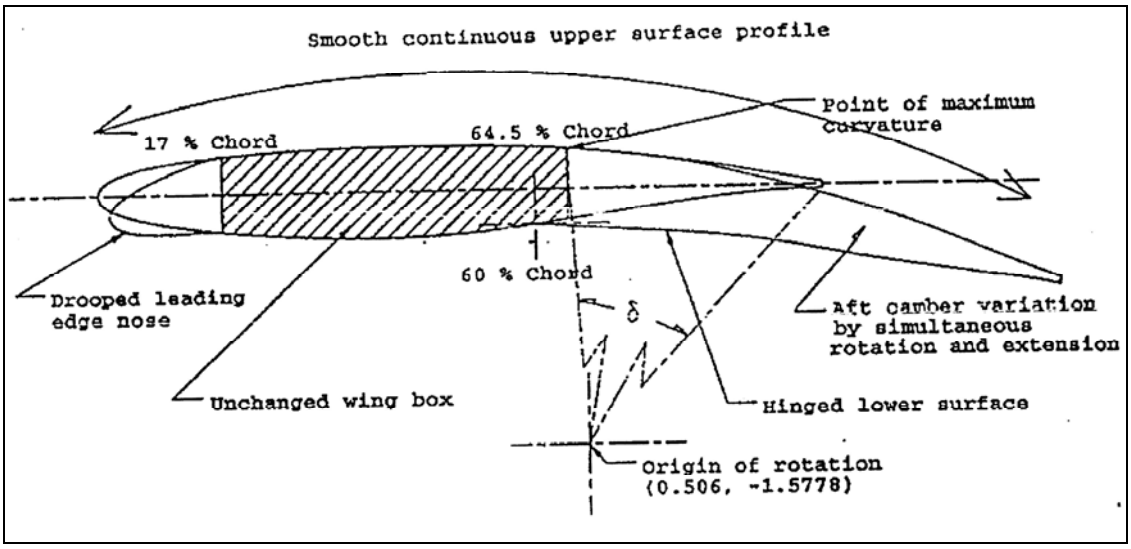

Figure 20 - Cranfield VC Concept [4]

This concept used a droop nose LE and a TE with simultaneous rotation and extension.

2.3.11 RAEVAM

The Royal Aircraft Establishment Variable Camber Mechanism (also called RAEVAM) was designed in the 1970's and consists of a flexible plate which is constrained by a series of swinging links attached to a rigid plate fixed to the spar (Figure 21). This concept is used as a droop nose design but also had several variations of the initial design, including chord extension[3] (Figure 22).

The RAEVAM concept with chord extension uses a rigid plate which sits in a track and is made to translate by a separate jack.
This design uses a flexible upper and lower skin to keep continuity, but there is also one case using a rigid lower skin (Figure 21).

As with the Boeing VC LE, this solution is complex and it is likely that the cost of maintenance will increase compared to a simpler design solution.

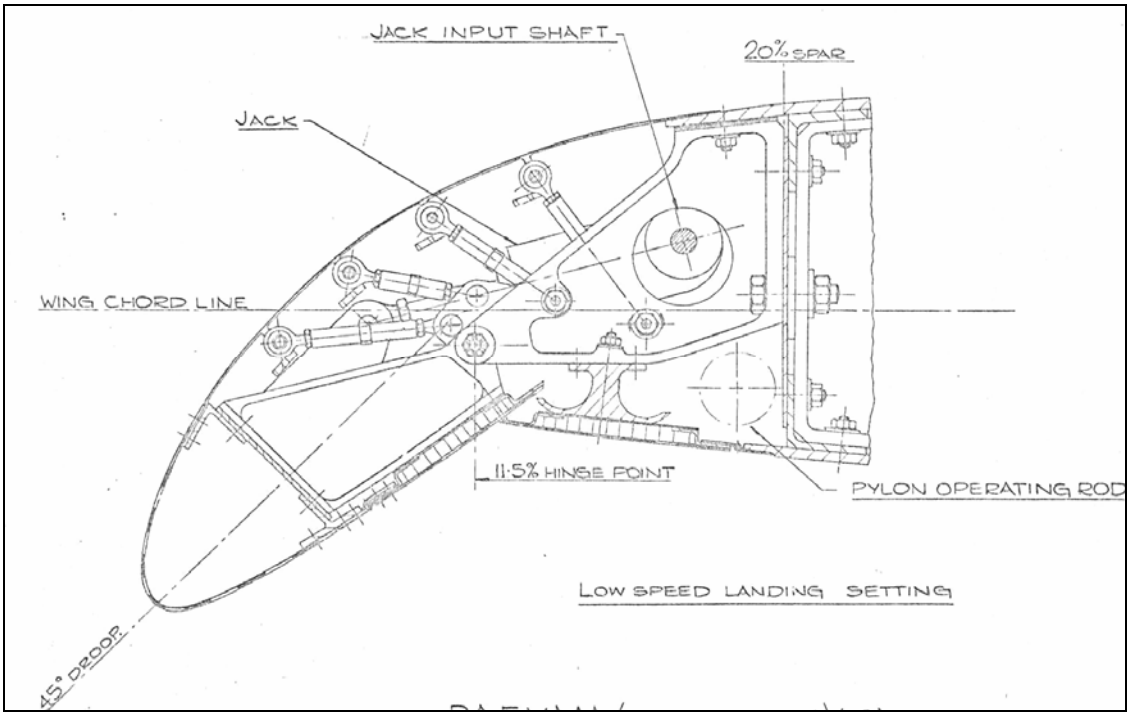

Figure 21 - RAEVAM design [3]

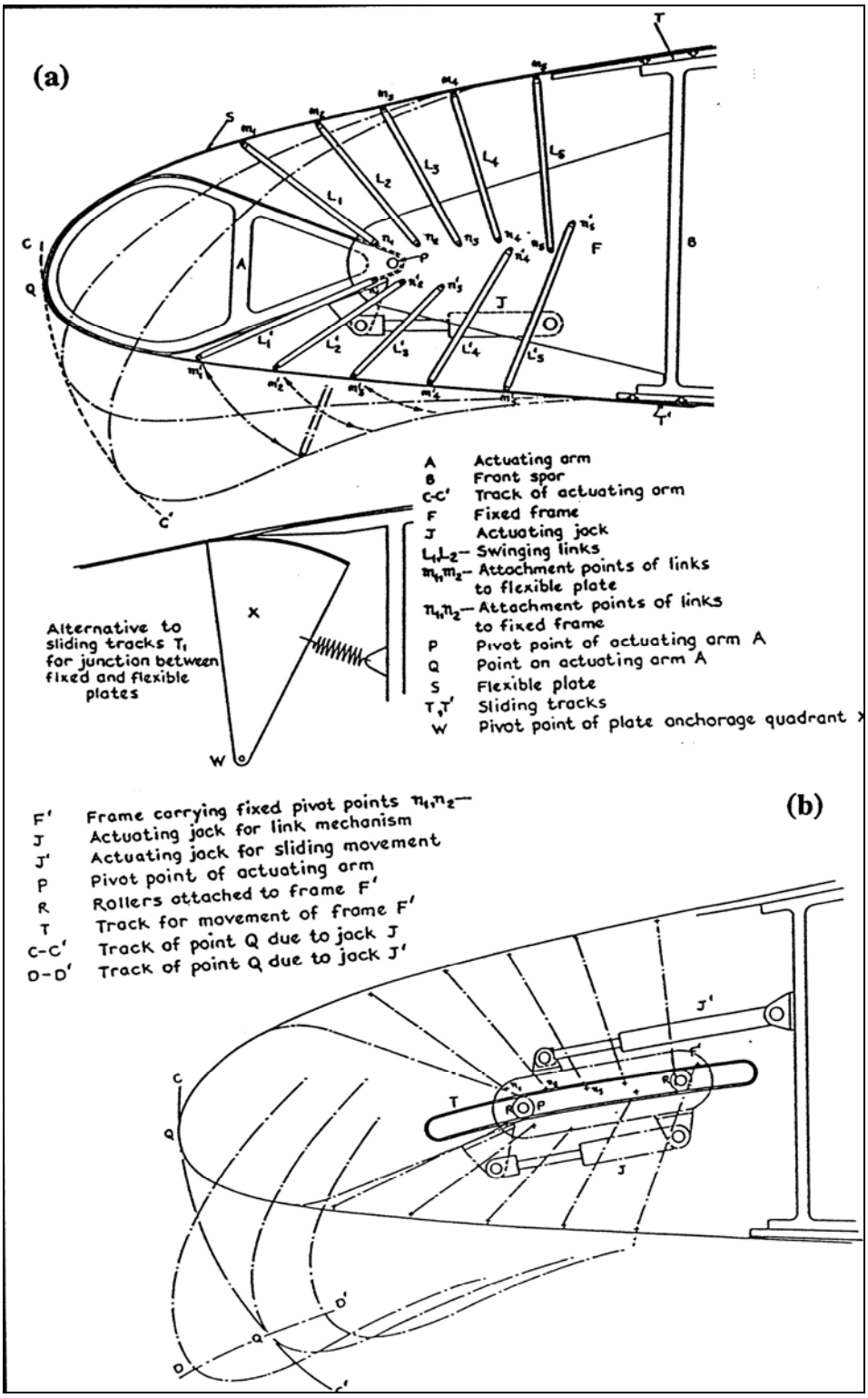

Figure 22 - RAEVAM [3]

2.3.12 The Banana tube mechanism concept

The "banana tube" mechanism gets its name from the banana shape of the rod. This mechanism is in fact quite simply a mechanism to generate a variable camber at the leading edge. The banana tube is rotated one way and then the airfoil (with a flexible skin) takes the shape of the rod inside, and so has a variable camber capability (Figure 23).

This concept is quite difficult to apply to a real aircraft since the rotation of the rod implies that there is translation of the end of the rod along the span of the wing. This concept needs a flexible skin, with bearing sliding along the span, and panels inside the LE, to capture the exact rod profile.

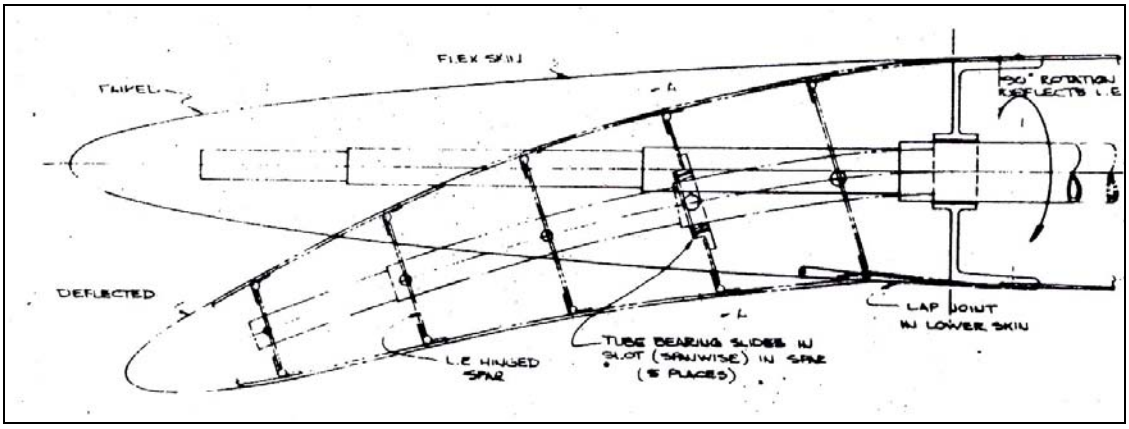

Figure 23 - "Banana tube" concept [25]

The same principle was tested by Daimler Benz at the end of the 90's (Figure 24), but was applied to the trailing edge [26]. It showed that the complexity of this concept is located inside the wing (or flap in this case) as there is a need for specific holes or gap within the wing (or flap) to allow the tube to move and so to give the exact shape to the wing. This means that the structural design of this wing would be quite complicated and needs further research and development.

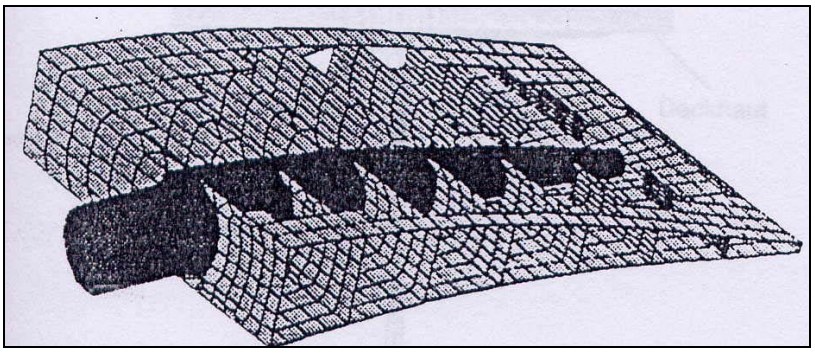

Figure 24 - "Banana tube" configuration [26]

2.3.13 New advanced technology VC wing

The LE mechanism for the Advanced Technology Variable Camber Wing (ATVCW) is shown below (Figure 25). It was studied in 1972 by Boeing and the Naval research department of Navy [25]. This design features the use of flexible skin panels on the upper surface of the wing. The forward edges of the flexible panels are attached to the nose beam while the aft edges are tied to the front spar cap. LE camber is increased by extending the actuator. This, in turn, rotates the actuator crank and actuator link causing the main support arm to rotate down. The main support arm rotations move the nose beam down, flexing the upper skin panel. A slave link connects the actuator link and a four bar linkage supported by the main support arm. The kinematics of this linkage are such that the nose beam is rotated the exact amount necessary to produce the required curvature of the flexible upper surface while also deflecting the LE downward[25].

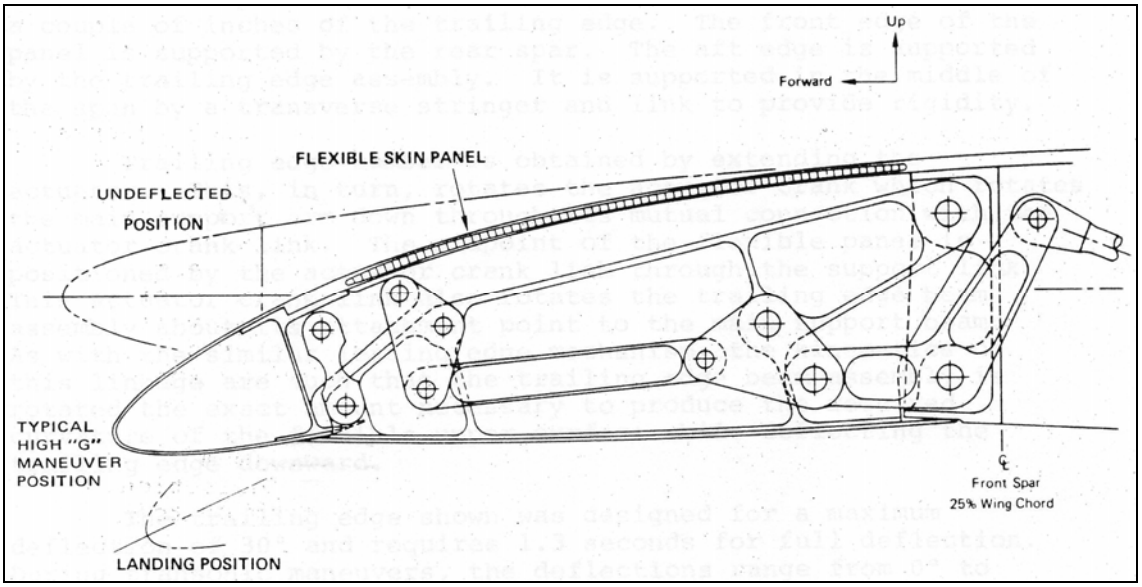

Figure 25 - Wing LE VC mechanism [25]

For this design the nose LE can deflect from 0° to 30° and from 0° to 15° during transonic manoeuvres.
The LE devices are separated into two panels on each wing, with one dual activator on each panel. These four LE VC mechanism (hydraulic) activators are controlled by a computer which deflects the LE depending on the Mach number. The computer sends signals to an electric control-servo which activates the actuators.

This concept was tested by Boeing using an F-8 as a test-bed aircraft. Boeing also took the opportunity to try different materials such as fibre glass and polyimide. Results showed that these materials are flexible as well as resistant. They were also shown to be less likely to propagate cracks, which is very important when considering bird strike or other objects striking the LE. However, due to the complexity of this concept (high number of parts) and complicated flight control systems, this concept was not used on other aircrafts.

2.3.14 Parker variable camber wing concept

The variable camber concept created by Parker [50] in the 1920's was revolutionary at that time and offered a solution to allow airfoils to camber in order to improve the aerodynamic performances of the wing. This concept was developed using the technology available at that time and so include a wood and metal structure enveloped by a fabric skin. The internal structure of the wing was made of channel section, tension links, reverse tension links and a tail piece. The main idea of this concept is that the fabric skin is attached to the different links of the internal structure, but this skin is allowed to slide around the front and rear spar. The tension links length will define the camber of the wing, as the length of these links will represent the kinematic and geometric characteristics of the airfoil. The spars are fixed and so the fabric and related structure can camber around the two spars (Figure 26). Only the tail piece is a rigid part which does not bend, it only follows the angle of camber of the skin at the rear spar. In the Parker concept the front spar is located close to the LE of the airfoil to offer a longer section of the airfoil to camber.

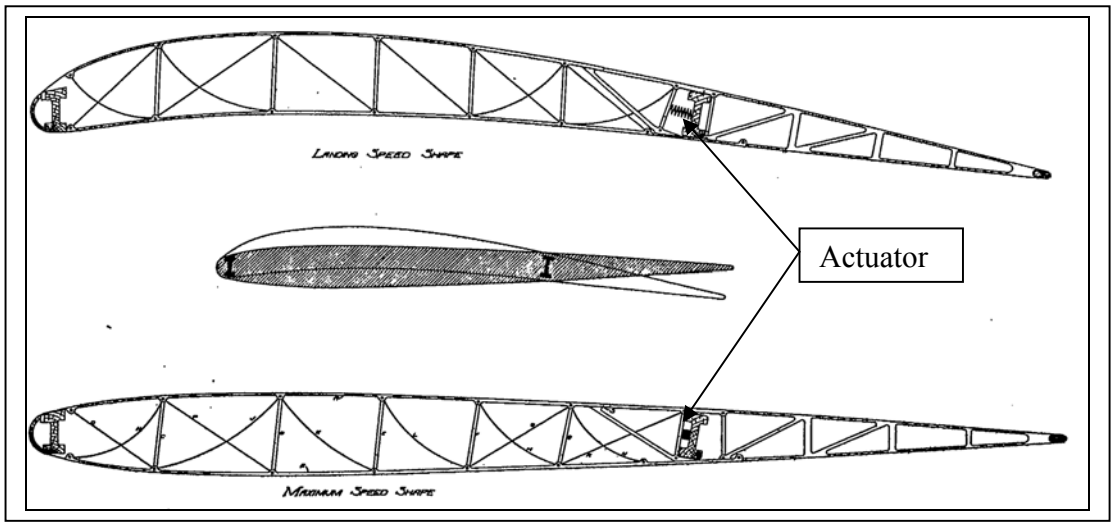

Figure 26 - Variable camber – Parker Concept [ref to NACA report]

This concept do not use VC LE device, but the author consider that similar concept could be potentially applied to LE devices. Also advances in technology by using composite materials for the skin and metallic or composite internal structure could help to apply this design concept to modern aircraft wings.

However, it has to be noted that this concept might generate sharp change of local curvature at the attachment points of the skin structure which in fact might lead to high stress concentration or discontinuity of the airfoil shape. These issues could prove detrimental to the performance of the airfoil.

2.3.15 Compliant mechanisms

Compliant mechanisms are mechanisms made of a rigid part having potential flexibility in some section in order to reproduce a more classical mechanism using rigid parts and joints (hinges or others). This type of mechanism can be applied to the design of LE and TE devices to achieve VC concept.

Flexsys engineers Kota and Hetrick developed a mission adaptive compliant mechanism for both LE and TE variable camber devices [51]. This concept uses compliant mechanism (see Figure 27) to change the shape of the LE of the airfoil by keeping a smooth profile even when full bend, with a claimed 6 Deg (see Figure 28) LE deployment 2which increases the lift of the airfoil by as much as 25%.

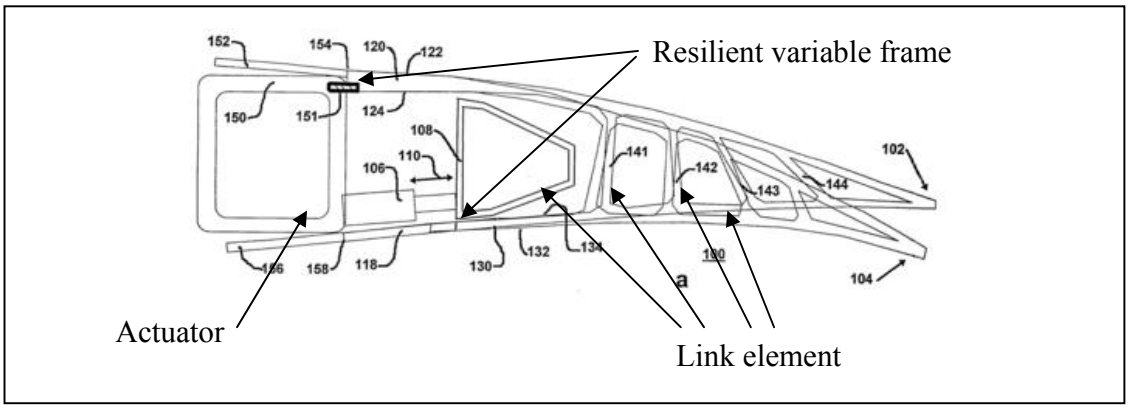

Figure 27 - Adaptive Compliant Wing and Rotor System [52]

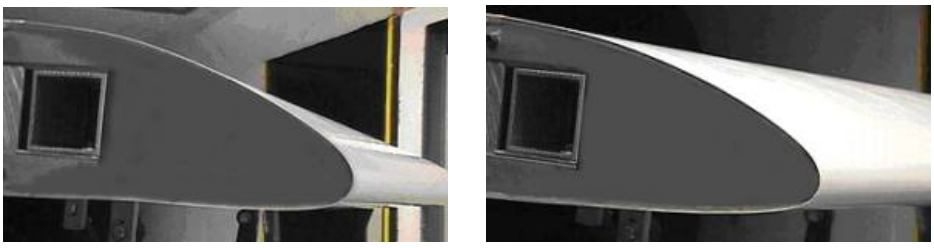

Figure 28 - LE Deflection of Compliant Mechanism [53]

This compliant mechanism is describe in Figure 29 could in fact offer a possible solution as it limits the number of parts and also avoid using hinges to deploy or camber the LE of the wing. This limit the assembly problems, the assembly cost and the maintenance compare to similar type of assemblies using hinges. Also it is planned for this type of compliant mechanism to be activated by only one actuator.

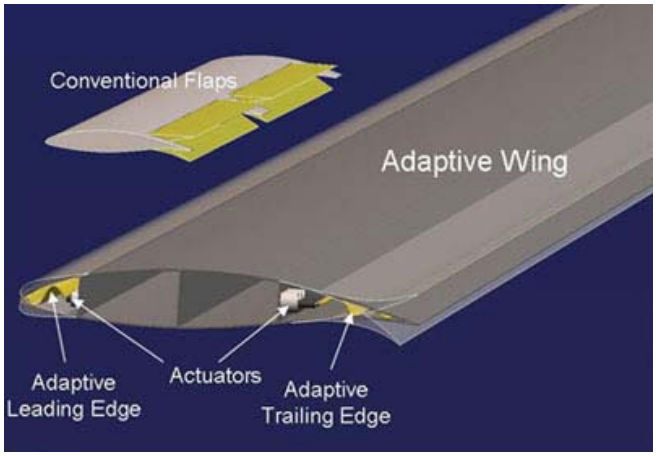

Figure 29 - Mission Adaptive Compliant Mechanism [53]

Similar type of truss structure (see Figure 30) have been investigated at Virginia Tech institute by M. Good [54] in order to design a compliant mechanism to use on aircraft tail using structural optimisation. Goode used ABS type of plastics (mainly made of rubber) to design his compliant mechanism, but this has shown limitations for manufacturing and use. The material used by Flexsys is unknown at present as they have not publicly shared the exact way their compliant mechanism is made and the material used.

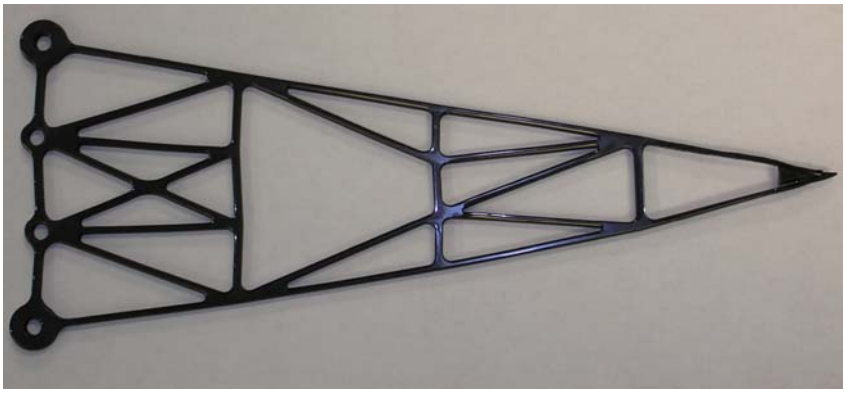

Figure 30 - Compliant mechanism by M. Good [54]

However, using this type of compliant structure do not allow for LE extension or possible gaps to re-energise the air. This type of concept could offer a possible "optimal" airfoil shape for LE and TE devices compare to the more classical "rigid " concept which only approximate the ideal airfoil profile. It also has to be noted that fatigue issues can occurs in these types of mechanisms and so this might be a concern during the certification process if used on commercial and military aircraft. Another downside of these types of concepts is the lack of fail safe opportunities as it is expected to certify the aircraft for safe fly. In fact, the compliant mechanism could fail with one of the structural member of the structure breaking, at which point the overall structure might behave in a totally condition both under aerodynamic loading and under the actuator loading. This could

prove a difficult point to overcome unless the overall structure is “designed” to completely collapse and so the other parallel spanwise structures will have to be submitted to higher loads, and these ones are designed to sustain the increase loading conditions.

These different solutions using compliant mechanisms have in fact a different effect than most of the previously described mechanism. These VC concepts are expected to be used during flight to help the pilot to flight the aircraft with limited use of the ailerons, it is expected that these type of devices will change the wing loading in flight to affect the trim performances of the aircraft and so give superior manoeuvre abilities to the aircraft. This mechanism could be used to improve the landing or take off performances but will have in fact limited effect as the general range of deployment is around 10 Deg. Also if these mechanism were pushed to go to higher angle of deployment, than the inherent design will probably generate sharp radii at the junction of the LE (or TE) to the wing box. This could be overcome by having higher degree of complexity in the truss structure by adding more flexible members, but that would both increase the weight of the overall system and make it more complex to manufacture and probably less reliable.

2.4. Technology review summary

This chapter described many concepts for LE device mechanisms, and also gave an insight on what technologies have been developed and used so far. The progression of the technology has helped the development of new mechanisms, but there is still a lot of "space" for investigation of new concepts and ideas using more modern technologies.
None of the radically new concepts are used on modern aircraft because the weight and complexity of the mechanisms to deploy the LE are a burden for these devices. No-one has yet found a way to design a mechanism which could be effective and at the same time light and easy to maintain and essentially cost effective. New concepts also have to provide the right aerodynamic performance. And it is necessary to have the perfect compromise between the performances and the cost of using these new concepts.
The time frames of when each concept has been developed show that little has been done on research for new LE devices since the 80's and 90's, or not many of them have been used on aircraft other than test-beds. The 21st century with new CAE tools, new materials and new manufacturing techniques should provide aerospace engineers with the necessary help to develop new solutions, and the remainder of this thesis will describe the author's effort to do so.

2.5. Design methodology review

Following on the review of the different types of technologies and concepts used for LE devices and VC wing solutions it is appropriate to consider different design methodology developed by other authors. The following paragraphs describe, briefly the different methodologies chosen to be analysed for this research as well as comments on the advantages and downsides of each one of them. Comparing the different types of methodologies will help to see the downfalls and possible improvements applicable to future design methodologies.

2.5.1 Ammoo VC design methodology

Ammoo design methodology [2] is based on a step scheme as there are no parallel processes, or concurrent engineering "activities", it is a one way process. This design methodology (see Figure 31 in page 31) provides an initial aerodynamic study followed by the mechanism and structural design stage.

Wind tunnel testing is done before the mechanical study as it only uses the ideal aerodynamic shape defined during the aerodynamic analysis.

The mechanical and structural section of the methodology provides the base for the design of the deployment mechanism and related sub structure.

Flight test follows on the mechanical study to evaluate the final performance of the integrated design solution; this is the result of the integration of both the aerodynamic and structural study.

This methodology is based on the choice of one concept for LE, or TE, during the aerodynamic study, however there is no consideration for cost or reliability optimisation. Only the mass parameter is considered during the structural analysis as users of the methodology will try to decrease the overall mass of the design solution. Also this methodology does not consider any type of kinematic chain synthesis or topology/dimensional optimisation and so does not supply the user with a possible tool to design the deployment mechanism. The user has to use a "trial and error" method to design a mechanism fitting the deployment trajectory.

This methodology provides, on the other hand, a detailed "route" to design specifically for LE and TE devices.

This design methodology does not provide a feedback or back loop between the aerodynamic and the structural study. This implies that this methodology does not allows the user to go back to the initial stage of the conceptual design to refine the solution in relation with constraints or problems discovered at a later stage of the process. In fact, possible changes of the profile due to mechanical constraints would not be linked into an aerodynamic analysis. Another downside of this methodology is the lack of help provided

to the user (estimation tools) at an early stage of the process to evaluate different solutions and the possible impacts and rough performances of different concepts. Also there are no mentions of improvement or optimisation phase of the weight, the reliability of the system or even cost, which in fact generates a result which is not at all optimised for any of these parameters.

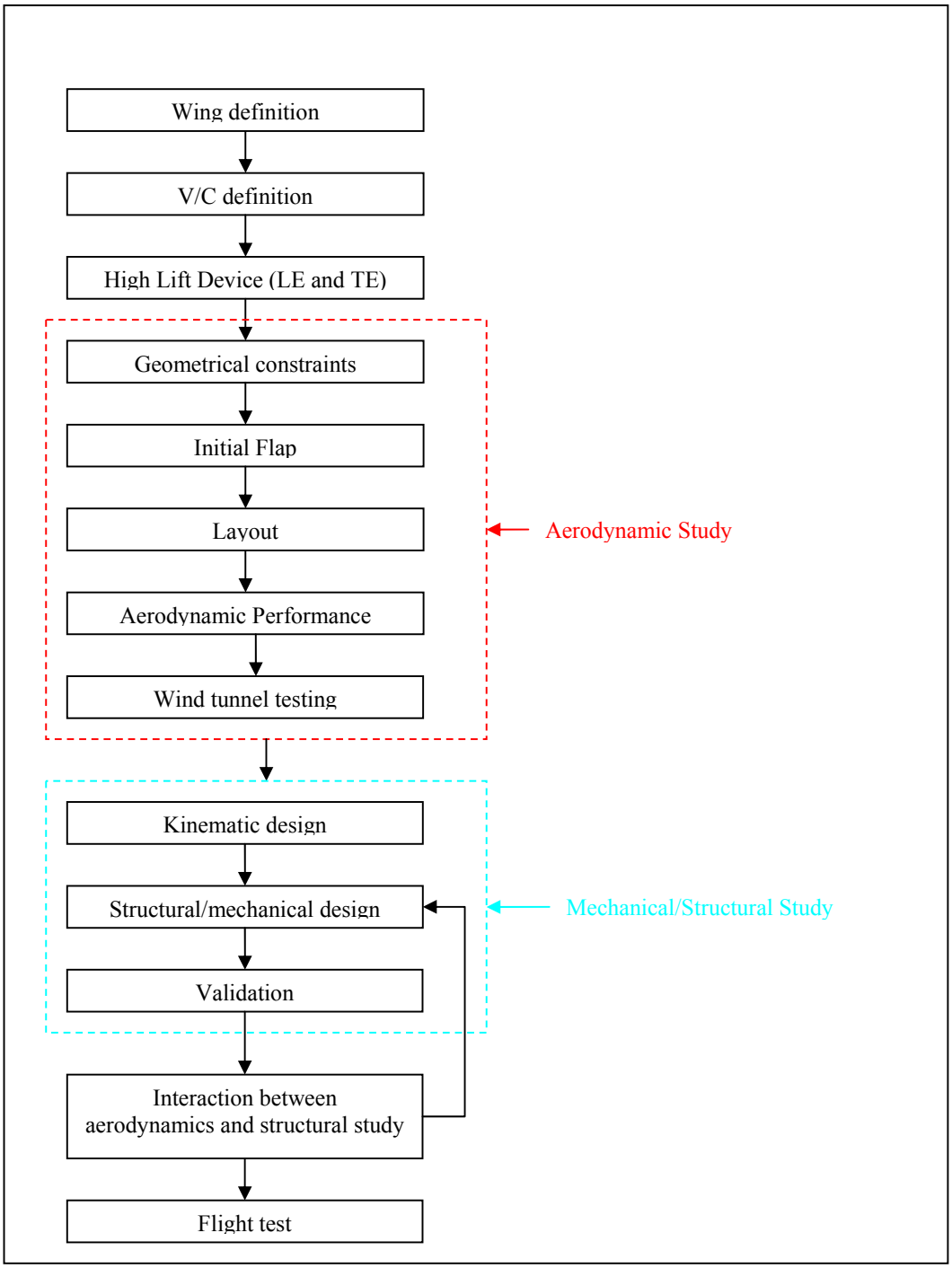

Figure 31 - Ammoo Variable Camber Design Methodology [2]

2.5.2 Pepper and Van Dam methodology

This methodology is a design process made around an initial chosen initial concept for high-lift devices, which is then improved. Pepper and Van Dam [6] process is starting from the initial chosen concept which is then followed by the initial layout (see Figure 32).

The following step is the "1st level design" which develops an initial optimal high lift configuration using specific aerodynamic tools and correlation formulas to obtain approximate cost and mass, as well as structural study. This gives the user an initial design solution with initial aerodynamic, noise, mass and structural performances.

This is followed by the 2nd level decision step which includes more detailed CFD analysis. This includes the more complete aircraft geometry including landing gear, engine and Centre of Gravity (CG) location.

This second level generates an optimum preliminary high lift configuration.

This is a two step optimisation process by reducing the mass, cost and noise as well as improving aerodynamic performances of the high-lift systems. This is a concurrent engineering process which can help to develop a solution quicker as the different teams can be working toward different part (aerodynamic, structure, noise…) of the chosen solution.

However, this methodology does not help the user to select initially an optimum concept for the mission requirement. Also this methodology does not generate mechanisms or kinematic chains for the user. This methodology only drives the improvement of a chosen initial concept. This methodology is based on a linear progression and improvement of a chosen design and offer little room for design innovation. This linear progression, on the other hand, will guarantee a regular improvement and will provide the user with a safe way to a potentially average design solution. Also it has to be noticed that this methodology do not consider VC devices at all, and so, only really consider the effect of the high lift devices mainly of take off and landing phases.

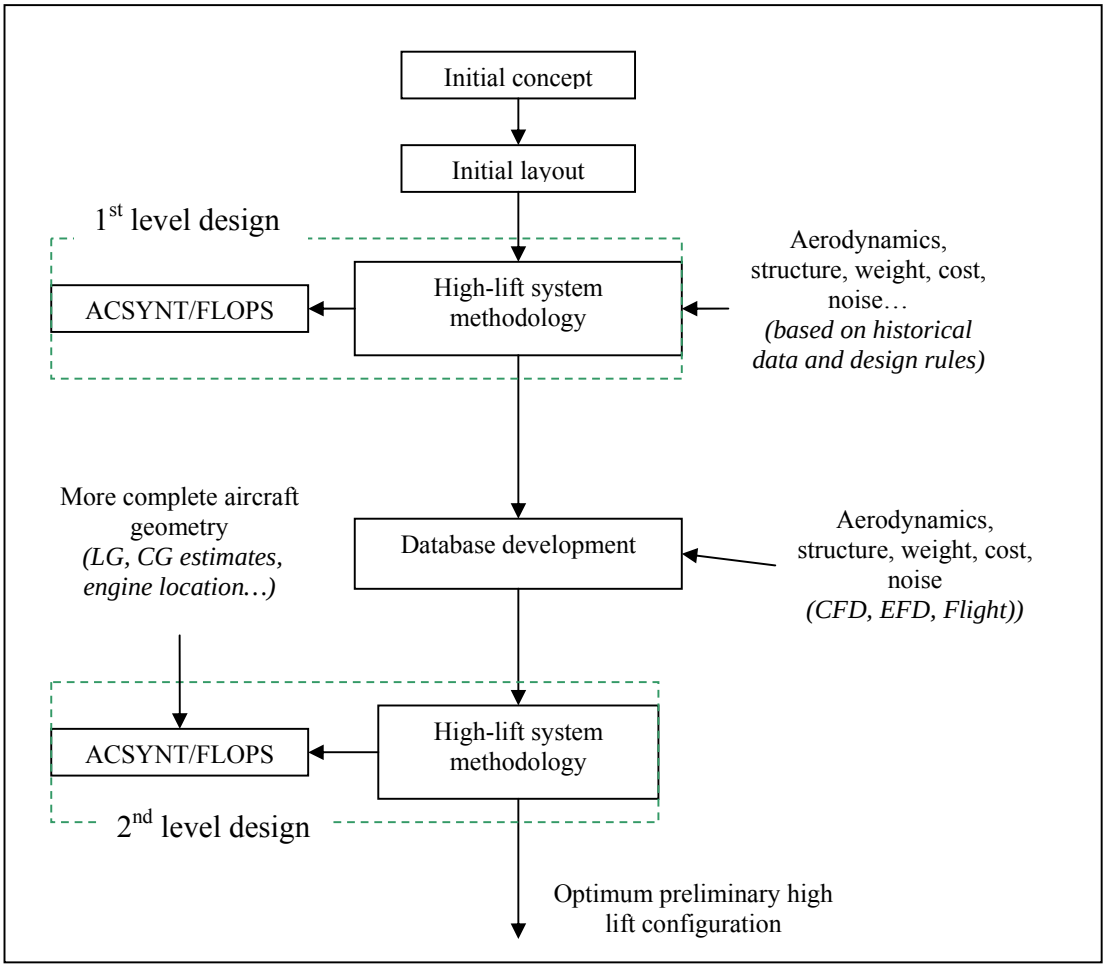

Figure 32 - High-lift Module (Pepper and Van Damme) [6]

2.5.3 High lift system design methodology (Flaig and Hillbig)

The methodology proposed by Flaig and Hillbig [55] include the study of the aerodynamic and also the structural and kinematics of the high lift systems (see Figure 33). This methodology can be applied to any high lift systems (LE or TE). One of the particularities of this methodology is the fact that is separates high speed case, for cruise, from the low speed case, for TO & landing. That fundamentally makes a difference from the other methodologies analysed in this report. The high speed case defines the wing design (wing layout and geometry) and the high lift devices are generated from this high speed case configuration.

This methodology follows a linear pattern, which also includes different feedbacks loops in order to optimise or improve the design solution to match the aerodynamic requirements. The overall process is broken down into three sections which are, the predevelopment phase, the development phase and the pre-flight phase each ones of these steps are described in more details below.

The Pre-development phase includes the estimation of initial aerodynamics performances of the wing. This wing is supplied from the high speed case wing. The authors also use specific estimation methods for the aerodynamic estimates. This phase also include an initial TO & landing performance check to see if the wing, or the aircraft, will have the initially required performances.

The development phase allows to push the design further and to improve the aerodynamics for the final high lift systems. In parallel to this comes the structural and kinematic study which provides the geometry for the deployment of the high lift systems. The result of the kinematics study is integrated within the aerodynamic analysis loop for the development phase, and once again the performance at TO & landing are checked and results are fed back into the analysis loop. The integration and both aerodynamics and structural study is completed during this phase.

At that point, there is a design freeze and most of the design parameters are fixed to build a prototype and go through the pre-flight test phase. This final phase includes final assessment of the TO & landing performances, but this time the result is not fed back to the aerodynamic analysis. This is a simple check to see if the estimates are in line with the flight test conditions.

During the whole process there is an aero database which is implemented and improved as the process goes from pre development phase to flight test in order to refine the quality of the final result.

This methodology does not include any optimisation or automatic generation of kinematic and structural solutions. That means the high lift deployment mechanism still have to be created using lengthy trial and error method. Another downside of this methodology is the lack of consideration for cost, mass and reliability of the solution

developed or to be developed. This could in fact lead to a good design solution to match the aerodynamic performance required but a poor solution for the mass and/or cost. Also there are no consideration of the potential use of VC concepts, which could prove to improve not only the low speed condition but also the high speed (cruise) condition.

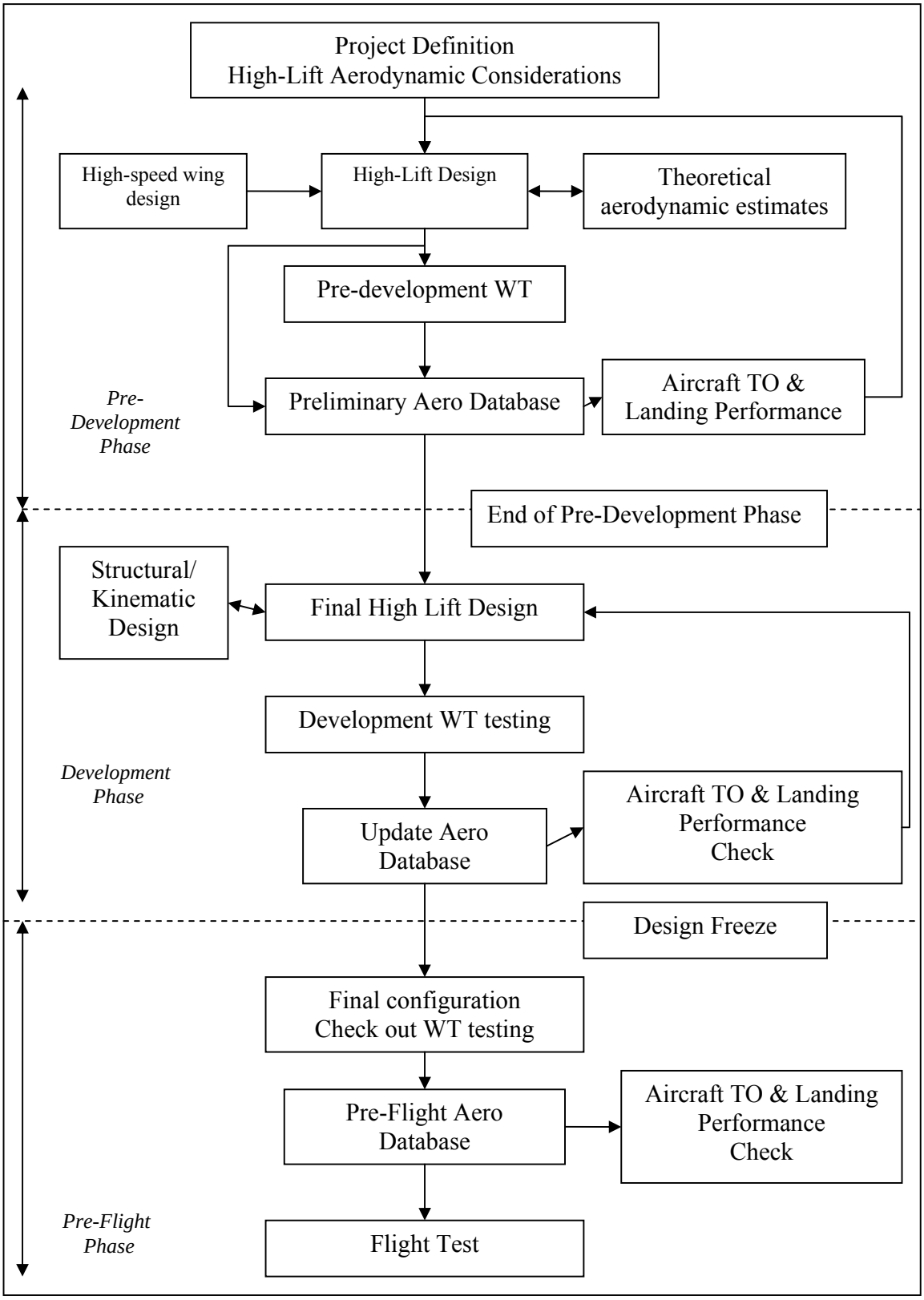

Figure 33 - Flaig and Hillbig High lift system design methodology [55]

2.5.4 Conclusion on design methodologies review:

The different methodologies presented in this chapter displayed various ways to approach the design of high lift devices (LE or TE) or VC devices, each one include different types of process and study or analyse different aspects of the LE and TE devices. However, by looking at the comments made by the author it is clear that there are room to improve further the design process of high lift devices and potentially the design of VC concepts. Also the presented methodologies, for most of them, do not focus (or include) VC solutions, and so do not offer the opportunity to use or investigate the use and effect of VC camber for LE devices. All of the methodologies presented do not offer any type of early estimations for different concepts, and so LE devices designers have to select or shortlist one or a few more classical concepts to run through the process to see if one would be better suited than the other. It also has to be noticed that only surrounding issues such as reliability, cost and weight are not considered in most of the design processes shown in the precedent pages. Another point to be added is that the design or generation of deployment mechanism (ie kinematic chain and associated structures) are not provided and so the user would have to use classical methods to investigate possible geometries.

Chapter 3

3. Methodology

3.1. Overall methodology – Design process

This chapter will describe the exact layout of the design methodology created to design LE device more efficiently and potentially using a VC option. The methodology developed for this research project will use the latest CAE tools to improve the time it takes to develop new design solutions. This methodology will also include the use of specific estimation tools (mass and reliability) developed specifically for this research project.

The new design methodology created (see Figure 34 page 41) begins by the identification of the main input parameters, which are the basis for the rest of the study. These parameters are the general description of the aircraft shape which includes wing planform, wing area, etc…These parameters are guided by the mission or flight requirements which will depend on the way the aircraft is flown. These requirements will help to decide on the major parameters for the wing design.

At that point the methodology graph splits into three sections, which are sections dedicated to the design of the three parts of the wing (LE, TE, wing box). However this research will only covers the LE design section. In order to clarify the overall wing design process and to show the integration of the LE design within the rest of the wing it was decided to show the whole process, even if some of this process will not be discussed in this report.

After the initial input parameters there is a split between mid-box, LE and TE design, the methodology (for this research) focus on the design of the LE of the wing.

To start the specific LE design process there is an early stage decision making for the mass, reliability, and cost estimation of the different kind of devices applied on the LE of the wing. New tools were developed by the author and used for this research, to assess different types of LE configurations, and to see their effect on the overall mass of the aircraft and reliability of the overall system. These new estimation tools were developped by the author and are explained and validated in the following pages. Also, it has to be noted that tese tools includes estimation for VC LE devices, which are in fact an innovation as other researchers have only estimated such parameter for classical devices only [15, 29, 30, 31, 32]. The results of these estimations help to choose a concept which might be a better solution taking into account the cost, maintenance and mass parameters.

Following these initial steps the methodology splits into two parts, which will work as a parallel process in order to design a fully integreated LE device. The structural and mechanical study is on one side, and the aerodynamic study on the other side. These parts are themselves made of several steps which will be described in the rest of this chapter.

The aerodynamic part of the methodology is used to analyse the effect a chosen kind of device for the leading edge on the overall aerodynamic performance of the wing. At a later stage this part of the methodology, is used to produce the loads needed to analyse

and check the structural integrity of the final design. These loads and set of pressure will be transferred to the structural part of the design process.

The mechanical and structural sides of the methodology focus on the design of a kinematic solution which complies with both the space available and with the structural requirements. This includes the design of the LE deployment mechanism and this also includes the design and analysis of the different component (FEA analysis).

The first step is the study of the mechanism design with mechanism synthesis followed by the dimension optimisation. The mechanism synthesis is done using the SYNAMEC software (developped by the SYNAMEC consortium [40]), which one the author used to include in the methodology. This new tool and its application will be described in the following pages as it provides the user with innovative way to design mechanism. However this is the first time that this tool has been used for the design of LE devices, it will also be used in this research to design VC LE devices.

Then this is followed by the structural study which allows analysing the size of the different elements of the mechanism in relation with the given loads.The data from the aerodynamic part of the methodology are used to be applied to the designed mechanism and the different parts of the LE device (LE skin, mechanism parts…).

The next "block", or step, is the interaction of the results from the parallel process of both aerodynamics and structural results. They are combined to see what emerges from the final design, and how to integrate the elements from both sides of the methodology.

There it is possible to assess the detailed mass and reliability of the system using the data from the mechanical and structural components of the methodology. This also includes the dimensions of the different components and the number of components.

The results of this design loop could then be sent back to an overall wing design multidisciplinary optimisation process that would feed back data to the beginning of the loop to improve the overall wing design.

The final step of the methodology is the manufacturing of a prototype, and flight test for the final validation of the design process results.

All of this is described in more detail in the following pages.

It has to be noted that the methodology diagram (ref to methodology diagram) shown the design process for the LE devices within the overall wing design process. This is to clarify to the reader the exact effect of the LE design compared to the overall wing process.

For the purpose of this study only the methodology "blocks" relative to the LE design will be studied.

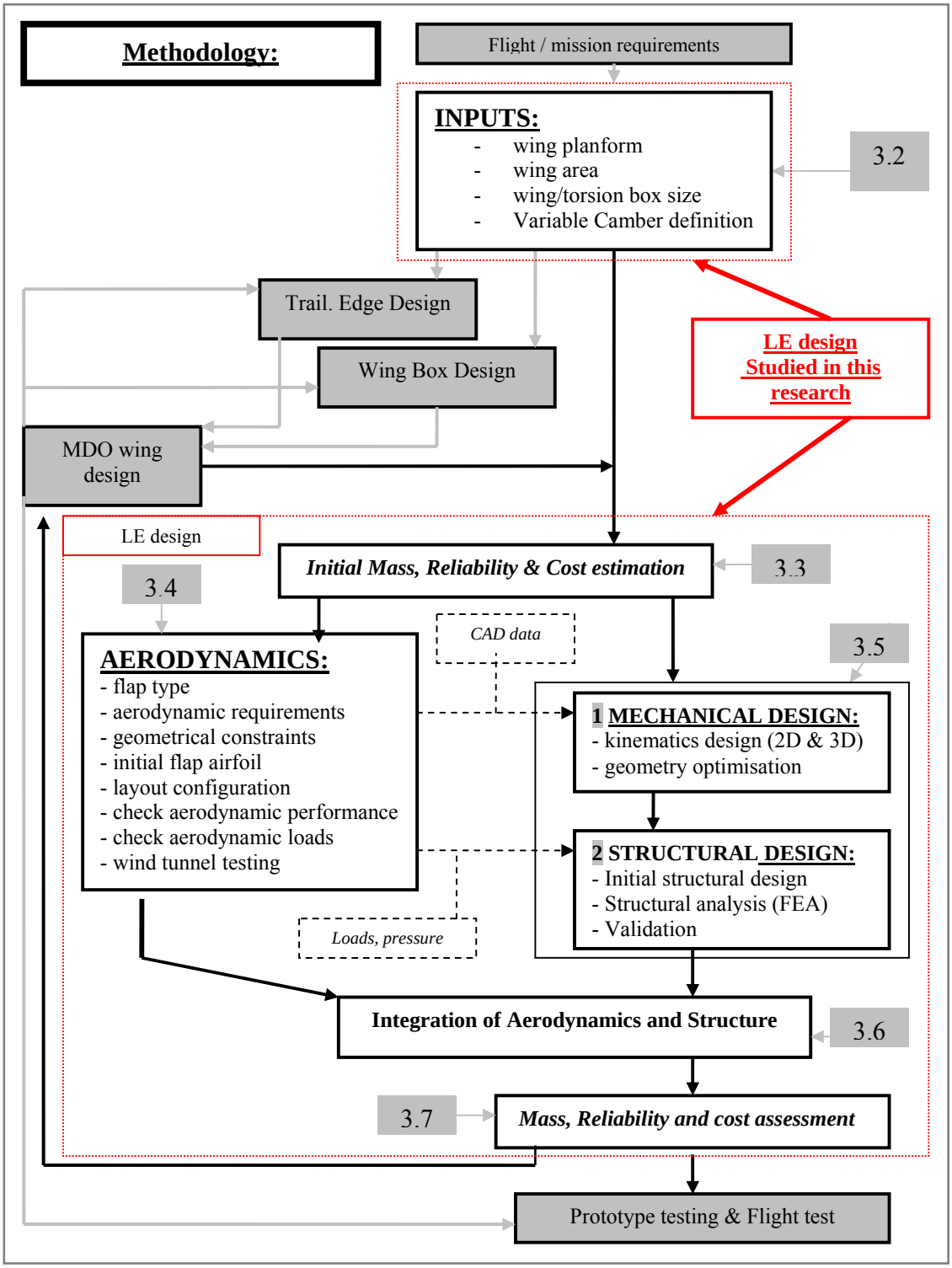

Figure 34 - Methodology diagram for the design of Le devices within the wing design process [Lajux]

3.2. Inputs

At the first stage of the aircraft design process it is essential that designers specify the basic inputs regarding the general aircraft parameters and requirements. These parameters will be used to design the airfoil and they are directly related to the general performance of the aircraft. It is normal that the wing is designed from scratch and that the general wing dimensions are decided, this is followed by more detailed design decisions regarding flap type or initial airfoil design.
In this part of the report there is a detailed description of the initial step of the methodology, with all the inputs required from the designer.

3.2.1 Wing planform

The wing planform is described by the general geometry and shape of the wing (Figure 35), which is the first parameter needed when designing an aircraft. These parameters are essential, as they will define how the aircraft will perform. They are also the basis for further design depending on what kind of aircraft is designed. Depending on the mission and requirements of the aircraft a totally different wing planform can be used.

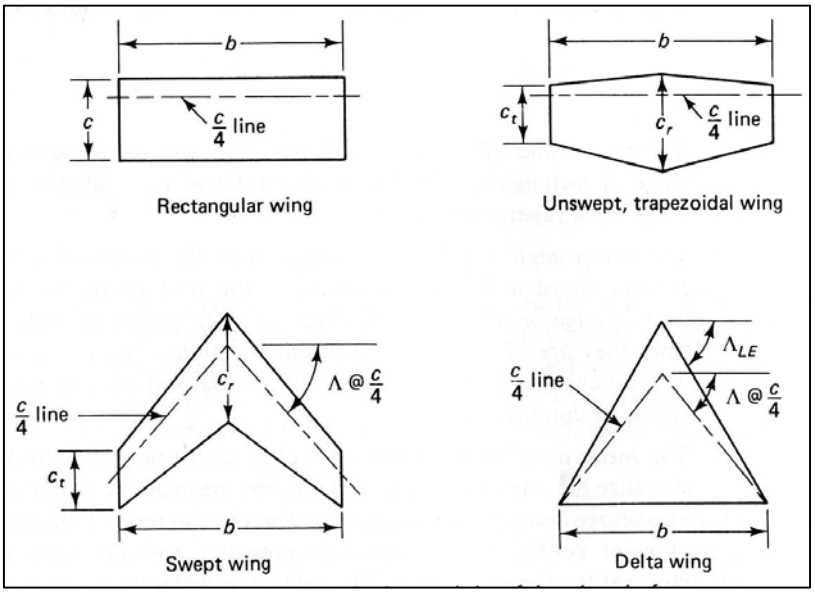

Figure 35 - Wing planform description [27]

The wing planform is described with the following parameters:

-sweep angle (Λ in [°])
-planform span (b in [m])
-Aspect Ratio (AR)
-Taper ratio (λ)
-Twist
-Thickness (optional at this stage)

With:

$$AR = \frac{b}{c} \text{ For rectangular wing;}$$

$$AR = \frac{b^2}{S} \text{ For non-rectangular wing}$$

Equation 3-1

AR= Aspect Ratio
b= Span [m]
c= Chord [m]

And:

$$\lambda = \frac{c_t}{c_r}$$

Equation 3-2

λ= Taper Ratio
c_t= Tip chord [m]
c_r= Root chord [m]

3.2.2 Wing area

The amount of wing area will be defined as a compromise between the design requirements and the performance objectives.
The wing area is defined as the plan surface of the wing, which is the product of the planform span (b) by the average chord ($\bar{c}$).

$$S = b \times \bar{c}$$

Equation 3-3

Where:
$\bar{c}$ = Average chord [m]
S= Wing area [m^2]

At this stage the design requirements for the wing can be defined using the payload to be lifted, the weight, the range and the cruise number requirements [28]. The payload and weight can be easily worked out using the lift equation.

The lift produced by the wing is relative to the speed of the airflow and the wing area (see Equation 3-4).

$$L = \frac{1}{2}\rho \times C_L \times V^2 \times A$$

Equation 3-4

Where:

L=	Total lift force	[N]	
C_L=	Lift coefficient	[--]	
ρ=	Density of air	[kg/m^3]	
V=	Free stream velocity	[m/s]	
A=	Area of the test element	[m^2]	(this is also the wing area = S)

The performance objectives of the wing design are the induced drag to reduce fuel burn, and the lift to increase Maximum Take-Off Weight (MTOW).

This design choice, or "compromise", is normally done using a chart to represent the different parameters compared to the design requirements. The parameters (wing, engines, empennage size, and weight) are normally scaled to meet the fixed payload and range as required, and DOC could also be added to define a more appropriate solution.

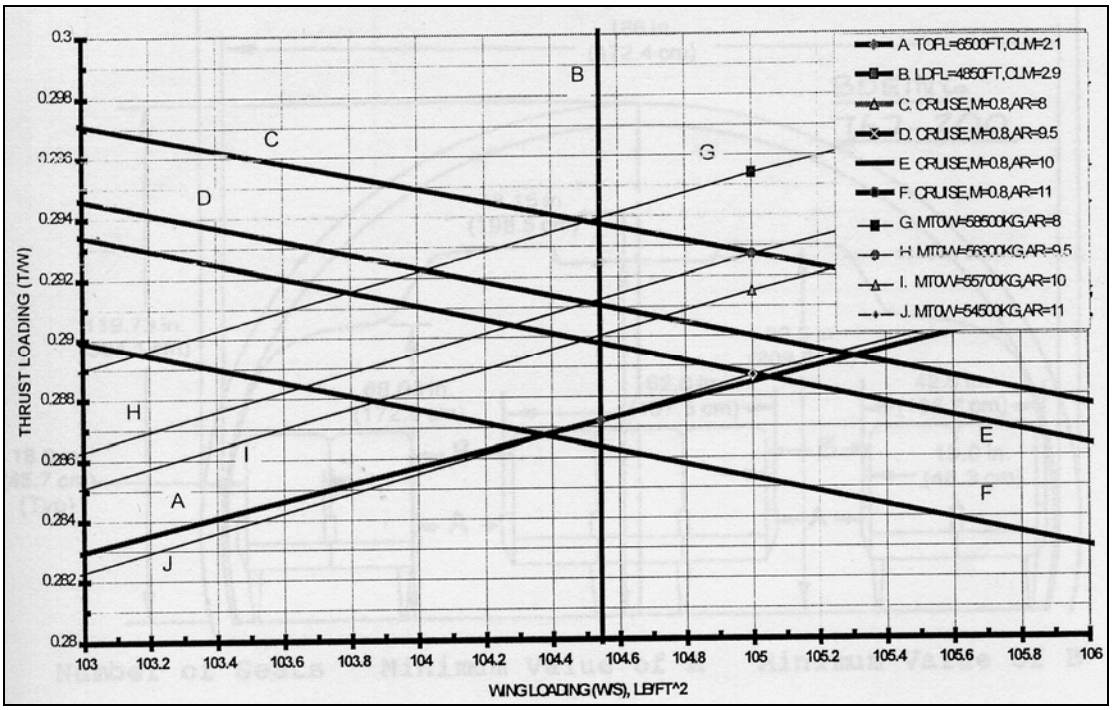

Figure 36 - ATRA-100 baseline wing and engine initial sizing [4]

The "design point" on the chart will be the optimum value for the wing area to satisfy the different requirements and objectives.
Using a different value for the wing area will either generate a too small area which means that there will be less volume for fuel and insufficient lift, or a too large area and then the weight will be high and cost will be excessive. Once the wing area is defined it is

possible to calculate the loading which acts directly to the wing structure through the applied pressure on the wing area.

An example of the change of wing area (stretching of the airbus A320 to A321) and its implications on design and cost can be found in Rudolph[10] (NASA research contract), even when this modification was done on the LE and TE device, the wing box remained unchanged.

3.2.3 Wing / Torsion box size

Being the aircraft's main lifting surface the wings have to carry the aerodynamic loads and the torsion box has to support the shear forces, bending moments and torques. The design of the wings is particularly important for the design of the LE devices as the packaging of these devices depends on available spaces in the wing. Aircraft wings usually use an assembly of spars and ribs to stiffen the wing and reduce weight using new materials with strong physical properties.

The spatial arrangement of the wing box and wing structure will define three things:

- space available to fit the LE device
- position of the LE spar where the LE device can be attached
- Maximum thickness

The above also apply for the trailing edge. Generally the position of the front spar is around 20% to 30% of the chord from the LE and 60% to 75% for the trailing edge. For the LE the generic 20 % of the chord is used to incorporate the LE device for the deployment as well as the associated substructure and systems.

3.2.4 Variable Camber definition

The reasons to use a VC device have been explained (see Chapter 2), but for the methodology the important point is that the wing geometry is changing. Using VC devices means that the planform shape and wing area are modified, and therefore improve the aerodynamic properties of the wing. This part of the methodology has slightly modified the data from the precedent steps and needs to be known for the following steps of the design process.

In some work done in the past at Cranfield University [3, 23] , the main focus was the effect of trailing edge device[4]. However, this research focuses on the effect of the LE devices on the wing performance. So the emphasis is on the effect of the LE device design on the overall aircraft performance.

When designing VC for LE device mechanisms there are several different aims which can be achieved, or have to be compromised [2]:

- Aerodynamic efficiency
- Reliability
- Low cost
- Low weight
- Low maintenance
- Icing

Most of the time, the cost will be the main point to be achieved as well as having a system which is easy to maintain.

Also at this stage of the design methodology, it is important to consider specific constraints linked to the LE device design. Indeed, it is of primary importance to establish the constraints for the aerodynamic expected from the design to be developed. Parameters to be considered are such as the maximum Cl for the wing in landing configuration and alternatively the Cl for take off and cruise conditions. This has a direct effect on the expectation from the variable camber profile generated at this step.

The maximum angle of deployment, used mainly for landing, should be 5 Deg for the "nose down" position. If the deployment is more than 5 Deg then the airfoil the Cl/Cd ratio will be detrimental to the airfoil performances. Also if any deployment angle for the nose down position used is more than 5 Deg then the implications on the design of the deployment mechanism will become more complex and then more costly and less reliable as well as heavier.

Same issue applies for the LE percentage of the overall chord length, it is expected that the LE device should cover around 20% of the chord at the front of the airfoil. Any values higher than 20% would reduce the size of the mid-box and so might weakened the wing structure. Although, it is potentially possible to consider a longer LE device for fighter aircraft, but for this research only commercial aircraft jet would be considered for more detailed design.

In relation to this it is expected that requirements are listed to define the average flight conditions, and the different characteristics the aircraft will have to comply with. These constraints will be the average landing distance, the maximum cruise speed, the maximum Cl (for landing). All these parameters once defined and fed to the rest of the methodology process will help to "drive" the design solution.

3.3. Mass, reliability & cost assessment

For any aircraft it is important to reduce the overall mass in order to carry more payload or fuel. The reliability is another important element to consider during the design process. Improving both the mass and the reliability results in designing a more efficient aircraft. So for this research a new set of estimation tools have been created. The mass assessment and the reliability assessment tools incorporate a table of weight and reliability estimation for LE device component sub-groups. This is done using a parametric method for one or more designs. These estimations are linked by the number and size of components in the system. It is easy to see that both estimation tools will work on the same format but in parallel, as they will be using the same kinds of data input. But this data will then be processed in a different manner for the different estimations. Also associated to this is a cost estimation using data from the both the mass and reliability estimations.

3.3.1 Mass assessment

There are two types of mass assessment, the first one is a preliminary mass estimation and, the second one is a detailed assessment which is performed once the final designs have been decided.

The preliminary mass estimation is used at an earlier stage in the methodology to make a decision about what kind of design to select in order to reduce the mass of the LE system compared with the aerodynamic performance and the overall aircraft mass [29].
The detailed mass assessment is done at the end of the design process once a virtual prototype (CAD model) has been designed. As all the components are designed in 3D it is possible to quickly find out the exact mass of each component knowing their volumes and densities. This is then summed to predict the overall system.

The Excel® program was used to create this preliminary mass estimation tool, and also other researches were referred relating to aircraft wing weight estimation such as the work done by Macci[30] [31] or the calculation methods given by Torenbeek[15] and Chatziliadis[32]. Some of these references have proven to be too simple to define accurately the LE device mass. So it was decided to use simple theoretical models by different authors and to improve on their work to add a more detailed analysis to obtain the projected mass as well as providing users with a quick solution. This resulted in the development of a fully parameterised mass estimation tool. This tool uses the same type of idea as the one described by Castagne[33] by using coefficient for the mass of each element or group of element instead of for the cost.

Both assessment methods are described in more detail in the following paragraphs (details of calculations can be found in the Appendix C (see page 162).

- **Preliminary mass estimation**

The preliminary mass estimation allows engineers to establish early in the design process what will be the general effect (in term of mass) of the different design solutions. This is more an estimation than a full assessment as it only generates projected mass results. This

is done using the general geometry of the aircraft and using mass ratio for components of each major component group of a LE device mechanism.

The preliminary mass estimation tool is useful to simulate several different possibilities (types of device or layout changes) and to help to choose the lightest option possible at the preliminary design stage.

Weight estimation for Leading edge devices
Leading Edge Description

Inboard Section (for one wing)				
Number of Panel=	3			
Span (each panel average)=	1.103	m	0.000	Ft
Chord for Section (average)=	1.158	m	0.000	Ft
Single Panel area =	1.277	m^2	13.748	Ft^2
Total panel area=	3.832	m^2	41.243	Ft^2
Inboard section span (overall)=	3.309	m	10.856	Ft

Type of Mechanism:	*Nb of panels*	*Mass*	
Slat with slave tracks	0	137.81	kg
Slat without slave tracks	0	79.70	kg
Fixed Camber Krueger flap	0	57.18	kg
Fixed Camb Krueger flap (bull nose)	0	111.28	kg
Variable camber Krueger flap	3	225.28	kg
Droop nose	0	97.32	kg
New variable camber concept	0	114.44	kg
Total mass (inboard)=		**675.85**	**kg**

Outboard Section (for one wing)				
Number of Panel=	6			
Span (each panel average)=	1.377	m	0.000	Ft
Chord for Section (average)=	0.317	m	0.000	Ft
Panel area=	0.437	m^2	4.702	Ft^2
Total panel area=	2.621	m^2	28.213	Ft^2
Inboard section span (overall)=	8.263	m	27.110	Ft

Type of Mechanism:	*Nb of panels*	*Mass*		*Mass*	
Slat with slave tracks	0	82.33	kg	181.50	lb
Slat without slave tracks	0	66.08	kg	145.68	lb
Fixed Camber Krueger flap	0	37.01	kg	81.58	lb
Fixed Camb Krueger flap (bull nose)	0	118.13	kg	260.42	lb
Variable camber Krueger flap	6	183.25	kg	404.00	lb
Droop nose	0	72.11	kg	158.97	lb
New variable camber concept	0	55.62	kg	122.62	lb
Total mass (outboard)=		**1099.52**	**kg**	**2,424.02**	**lb.**
Sub-total for one wing =		**1775.37**	**kg**	**3,914.03**	**lb.**
Full aircraft L.E. mass estimation=		**3550.75**	**kg**	**7,828.060**	**lb.**
Full aircraft L.E. area =		**6.453**	**m^2**	**69.459**	**Ft^2**

Table 1 - Example of the mass estimation "general layer" [Lajux]

The other advantage of this tool is the simplicity of use and its flexibility which helps to generate more accurate results, using different layers of this tool structure. This can be

done by changing some of the original parameters to represent more accurately a LE mechanism. There are two "layers" to this tool, the first layer to choose the type of device and the general layout of the aircraft ("general layer"), and the second layer enables the user to change the parameters used to represent each component in more detail ("component level"). Figure 37 shows the overall structure of this tool. More explanation on the structure, and the analytical scheme used for the calculation can be found in Appendix C (page 162).

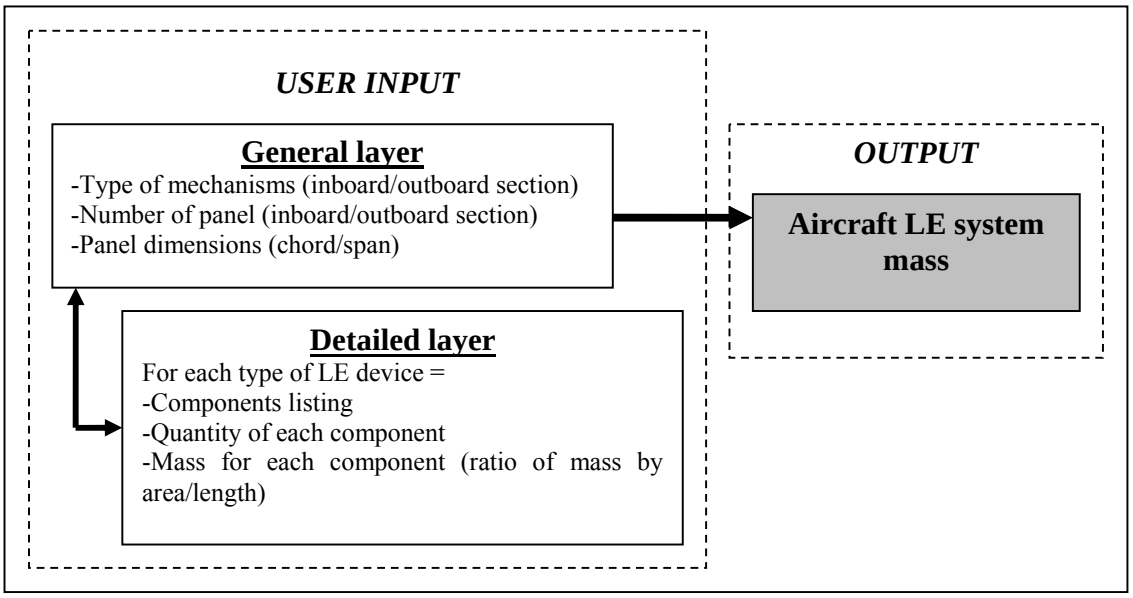

Figure 37 - Structure of the mass estimation tool [Lajux]

This initial mass estimation tool allows quick determination of the overall mass of a LE system using the general aircraft geometry (which can be changed). The estimation can be quickly generated and uses simple parameters representing the different component group's weight depending (see Equation 3-5 in page 50) on the chord and span of the LE devices.

Another option available, in the component layer of the program, is to change the ratio of mass/chord length (and span length) or to change the material (density) of each component sub-group. This assessment also takes into account the mass of the other smaller components of the system (this is a mass/projected area ratio), this is then summarised to give the mass for each panel and device type.

This tool also allows the user to see the effects of different kinds of LE device layout on the wing of the aircraft as well as the impact of changing the type of LE mechanism and the material used. The result of this process provides a good quality mass target for subsequent design phases.

The "component level" mass is calculated using the following equation. This equation defines the mass of each panel depending on different mass parameters (described below):

$$M_{COMP} = \alpha(M_{COMP,Specific} + M_A.PA + M_c.c_{average})$$

Equation 3-5

With:

α=	Quantity of component	[--]
$M_{COMP,Specific}$=	Specific mass of component	[kg]
M_A=	Specific mass ratio per unit area	[kg/m^2]
PA=	Projected panel area	[m$^{2]}$]
M_C=	Specific mass ratio per chord unit	[kg/m]
$c_{average}$=	Average panel chord	[m]

The different specific mass ratio per area have be defined by the author following an extensive survey of the mass for different components on current aircraft related to the area of the related panels and the numbers of actuators per panel [41,42,43]. These parameters could be refined by the users following the mass assessment at the end of the design process to further improve the accuracy of the mass estimation. Has technology evolve and structural optimisation regularly improves it is expected the M_A value will decrease in the future.

Mechanism description for Inboard section for one panel=

Slat with slave tracks

Elements	Nb		mass/comp	mass/A-L (kg)	Panel ch (m)	Panel A (m2)		Mass (kg)
Slat	1			20.00		1.9157952		38.32
Main tracks	2			15.00	1.158			34.74
Main rollers/ma	4		3.00					24.00
Side rollers/mai	4		2.00					16.00
Slave tracks	2			8.00	1.158			18.53
Slave rollers/sla	2		2.00					8.00
Actuator/main t	2		2.00					8.00
Miscellaneous	1		3.00					3.00
							TOTAL=	150.58

Table 2 - Example of the mass estimation "component layer" [Lajux]

⇨ Detailed mass assessment

The detailed mass assessment can be done at the end of the preliminary design stage, or, detail design stage process to see exactly what the overall mass of the LE systems is. But more interestingly the mass estimation tool can be used to compare the results with the planned mass from the preliminary mass estimation and see what the difference is and check the accuracy of the early decision making tool.

This final detailed assessment is done using the CAD model of the LE system and introducing material density to represent the different materials used in the LE device. This assessment also helps to see where most of the weight is located, as this might prove important to know, especially when doing some dynamic analysis of the deployment of the LE device or the inertia and Centre of Gravity (CG) of the aircraft.

However this is not much help during the design process as it does not help the designers to make a choice on what option to choose.

3.3.2 Reliability assessment

Reliability must be considered early in the design process when changes are most easily and economically made[34] as it helps to get a more efficient final solutions, and lower operational cost. The tool which has been developed for this research helps designers to choose the best design solution and also helps to see what will be the possible induced cost of using different designs, as the reliability will have a direct link to the operational cost. Once the detailed design stage is completed, it would be costly to modify the chosen design due to the large amount of resources already committed [35] whereas taking into account the effect of reliability at an early stage will reduce the cost of the possible modification at a later stage. The reliability is a part of the design methodology as it helps to get to the optimum design quicker.

The tool developed to assess and estimate the reliability of the different LE devices has been designed in a similar way to the mass estimation tool; it has two levels as there are two types of evaluations possible. This reliability tool is based on the part count reliability prediction, where the failure rate of each components are summed to give the overall reliability [36]. The main reliability calculation theory used for this tool is the part count has the author relied on the estimation of failure rate for the number of parts, but it also use a block calculation theory to calculate the failure rate of assemblies [34], more detail on the exact calculation can be found in Appendix C (see page 166). The reliability estimation is based on average failure rates for parts, or sub assemblies.

The first type is for the early design stage decision making and also sees the general impact of different design solutions on the aircraft. The second type is for the final detailed assessment of the chosen design. It helps to give the accurate assessment of the system and sub-system reliability.

⇨ Preliminary reliability estimation

The preliminary reliability estimation allows engineers to establish at an early stage in the design process what will be the impact of choosing one design compared to others, in term of reliability. This is more an estimation tool than a tool to produce a refined solution. It is used to get the overall reliability projected result, using the aircraft main geometry and the way the aircraft is used.

A complete new tool has been developed to generate the estimated result for the reliability estimation of different types of LE devices [29]. Each type of device has been

represented to be able to generate the reliability estimation for each of them depending on the aircraft LE system and the aircraft use. It was decided to use the general failure rate for each component of a LE system in order to generate the overall system reliability. The overall system reliability is calculated by adding the specific failure rate of the different LE sections (use of the part count method[36]), this is described in Equation 3-6 et Equation 3-7. In order to do so general reliability references have been used [34, 37] as well as using reliability parameters and values used by aircraft manufactures[35].

At the "component level", the equation gives the reliability for 1 or more component of the same kind:

$$F_{COMP} = \alpha(F_{COMP,cycle} + F_{COMP,time})$$

Equation 3-6

F_{COMP} =	Component failure rate	[per 10^6h]
α =	Number of component of each type	[--]
$F_{COMP,cycle}$ =	Specific component failure rate	[per 10^6cycle]
$F_{COMP,time}$ =	Specific component failure rate	[per 10^6h]

The specific component failure rate can be estimated by a number of cycle (for parts like actuators being used at take off and landing), or by a time (for parts such as flaps).

All of these equations are based on the fact that a system failure rate (Sigma s) is equal to the summation of the sub-system failure rates (Sigma l) as mentioned by Dr M. Bineid in his PhD thesis[35] .

$$\lambda_s = \sum_i^n \lambda_i$$

Equation 3-7

The advantage of this tool is that it is really easy to use at the decision-making stage. It gives immediate values for the reliability of the different types of devices, and also gives a value for the reliability of the overall aircraft LE systems. In fact it is possible to have a combination of different devices along the wing and still get an overall result and a result for each device (see Table 3). This is really helpful to assess the effect of the different type of devices used along the wing. This tool is designed using the Excel® software to create a kind of parametric matrix, with the preliminary reliability estimation being the first page (Table 3) which compiles all the results to give a general idea of the reliability for each type of device. It was decided to develop this tool in such a way that it is possible to see the effect of each device on the inboard, outboard and central section of the wing, and it is also possible to have a combination of devices on each section.
More details and explanation on the tool structure and analytical schemes used to generate the result can be found in Appendix C.

Central Section (for one wing)			
Type of Mechanism for the panel described above		Failure rate (10e6 h)	Failure rate (h)
Slat with slave tracks	0	643.746	6.44E-04
Slat without slave tracks	0	585.344	5.85E-04
Fixed Camber Krueger flap	0	205.876	2.06E-04
Fixed Camb Krueger flap (bull nose)	0	197.636	1.98E-04
Variable camber Krueger flap	0	464.045	4.64E-04
Droop nose	0	198.808	1.99E-04
New variable camber concept	0	221.747	2.22E-04
Total failure rate=		0.000	0.00E+00
Outboard Section (for one wing)			
Type of Mechanism for the panel described above		Failure rate (10e6 h)	Failure rate (h)
Slat with slave tracks	0	643.746	6.44E-04
Slat without slave tracks	0	585.344	5.85E-04
Fixed Camber Krueger flap	4	205.876	2.06E-04
Fixed Camb Krueger flap (bull nose)	0	197.636	1.98E-04
Variable camber Krueger flap	0	464.045	4.64E-04
Droop nose	0	198.808	1.99E-04
New variable camber concept	0	221.748	2.22E-04
Total failure rate =		823.504	8.24E-04
		Failure rate (10e6 h)	Failure rate (h)
Total failure rate/wing=		1.221E+03	1.22E-03
Total failure rate /aircraft=		2.442E+03	2.44E-03

Table 3 - Example of the reliability assessment tool at "aircraft level" [Lajux]

⇨ Detailed reliability assessment

The same tool is to be used at a later stage of the design process to see the detailed effects of the chosen solution. In fact it is possible for the user to access the second level of the program (Table 4). This level is detailed enough to generate an accurate result of the system reliability. It is also possible to adapt this program to match the exact design solution and have a more accurate result.

This second level, which is largely more accurate than the first level, represents each part of a LE system, and breaks it down to each sub component. Each component is then allocated a value for its failure rate and the result generated reproduces the exact reliability of the system.

Both estimation tools will be used at an early design stage for decision making regarding the type of device to select, depending on the wing planform and the way to use the aircraft.

Mechanism reliability description for Inboard section for one panel=					
Slat with slave tracks					
Elements	Number		failure rate/cycle (10e6cycles)	failure rate/element (10e6h)	sub-sytem failure rate (10e6h)
Slat	1			0.001	0.001
Main tracks	2		0.001		0.002
Main rollers/main track	4		14.600		116.800
Side rollers/main track	4		14.600		116.800
Slave tracks	2		0.001		0.002
Slave rollers/slave track	2		0.146		0.584
Actuator/main track	2		87.935		351.740
Miscellaneous	1		0.001		0.001
				Assembly failure rate =	585.930

Table 4 - Example of the detailed layer of the reliability assessment [Lajux]

Finally these two tools can be used at the end of the design process to produce a final accurate view on the overall mass and reliability of the system.
The difference of mass for the different LE devices has a great influence on the overall DOC of flying the aircraft. In fact, carrying extra mass on an aircraft means that with the same amount of fuel the length of each trip will be reduced, or, the payload will have to decrease. Either way, this will impact the profit margin of the airlines.

3.3.3 Cost assessment

Following on the analysis of the mass and reliability estimation it is important to run a cost analysis for the use of the different LE devices. This cost analysis gives an idea of the cost involved with "making" and "running" the different devices. This cost is of a prime importance to the airlines as they will be able to see what financial benefit each type of device will bring.

The cost assessment module is separated in different sections which will help to analyse the cost of each device from a different point of view.
This analysis tool include a maintenance associated cost module (plus its own fleet cost management add on), a manufacturing cost estimation, and finally, a mass cost estimation.
These three sub-modules of the cost estimation tool are described in the following paragraphs.

⇨ Manufacturing cost

The manufacturing cost module includes generic cost estimation for the manufacturing of the different types of LE devices based mainly on the average part number and dimension of the LE devices section span. The initial cost of producing the part also has its own importance as well as the possible material choice for the LE devices. This module uses some of the inputs data to define the average cost for each configuration. Using these data and some others coming from the mass estimation allows users to define the cost of making the part. To calculate more accurately the manufacturing cost the author used a specific cost to mass ratio for different types of materials commonly used in the aerospace industry. The author also included a "buy to fly" ratio to define the amount of material lost between the purchase of the original material and the mass of this material flying in the aircraft. In fact, depending and the material used and the manufacturing techniques, the overall loss of material can be quite significant and have a major influence on the overall price of each LE device assembly (see Appendix C in page 169).

This allows the users to improve further the assessment of cost for making each part. Also this tool has imbedded advices and comments for the user to refine the results depending on the type of material selected.

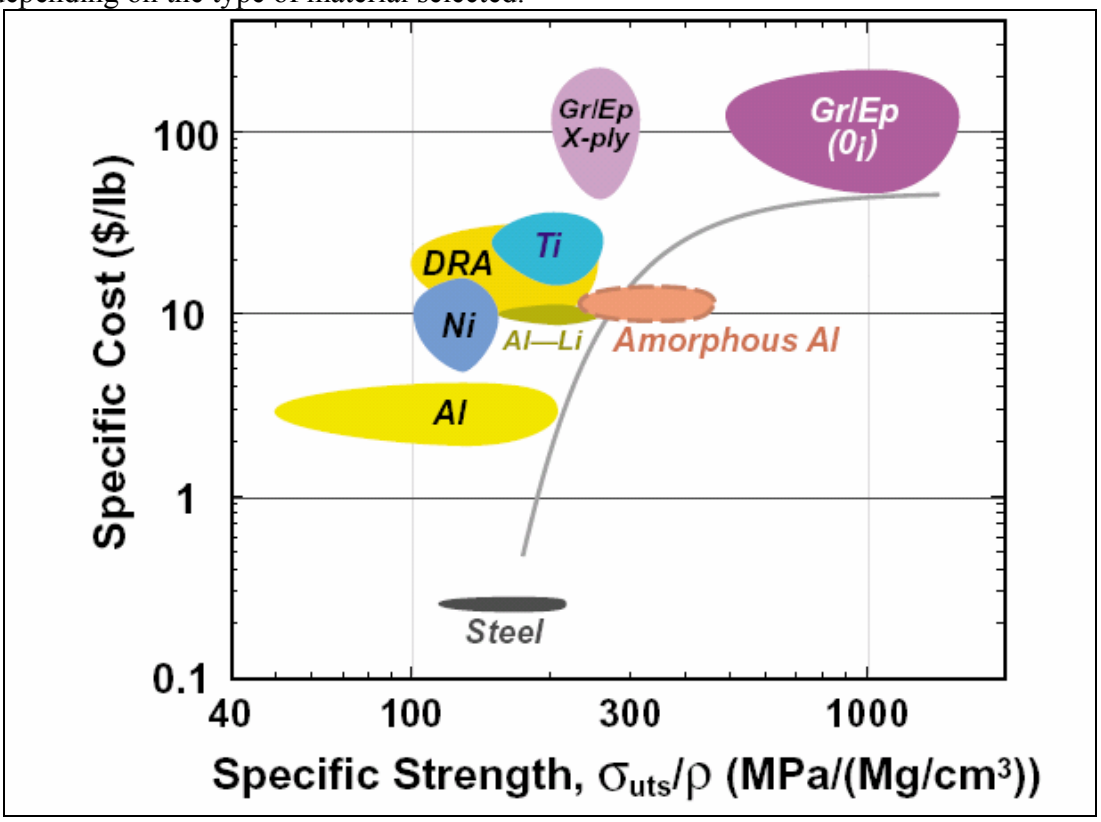

Figure 38 - Specific cost of material[48]

Depending on what technology to use the cost might be greater, however this could be linked to a potential mass reduction and so a possible cost saving while flying the aircraft, this is analysed in the next paragraph.

⇨ Maintenance associated cost and fleet management

It is possible to analyse the exact cost of "running" each type of LE devices through the maintenance cost module. This allows to calculate the cost of the maintenance (with data from the reliability module) to see what cost will occur with the maintenance of the devices. This module also incorporates an analysis of the possible fleet management to discover what the cost is to different airlines to use different types of devices in different conditions.

The maintenance associated is calculated on the basis of the reliability data from the previous paragraphs, the amount of failures for the different parts will represent the frequency of the maintenance checks and changes of parts. Then for the different devices depending on the average part numbers a generic cost is estimated for the maintenance check and for the amount of time lost for the airlines. This way, it is possible to evaluate different configurations depending on the type of devices and the number of panels and sections for the wing studied. Also by reducing the maintenance cost the airlines can

decrease the DOC, as seen in the following graph the overall cost of maintenance for a regional jet represent more than 10% of the DOC see Figure 39).

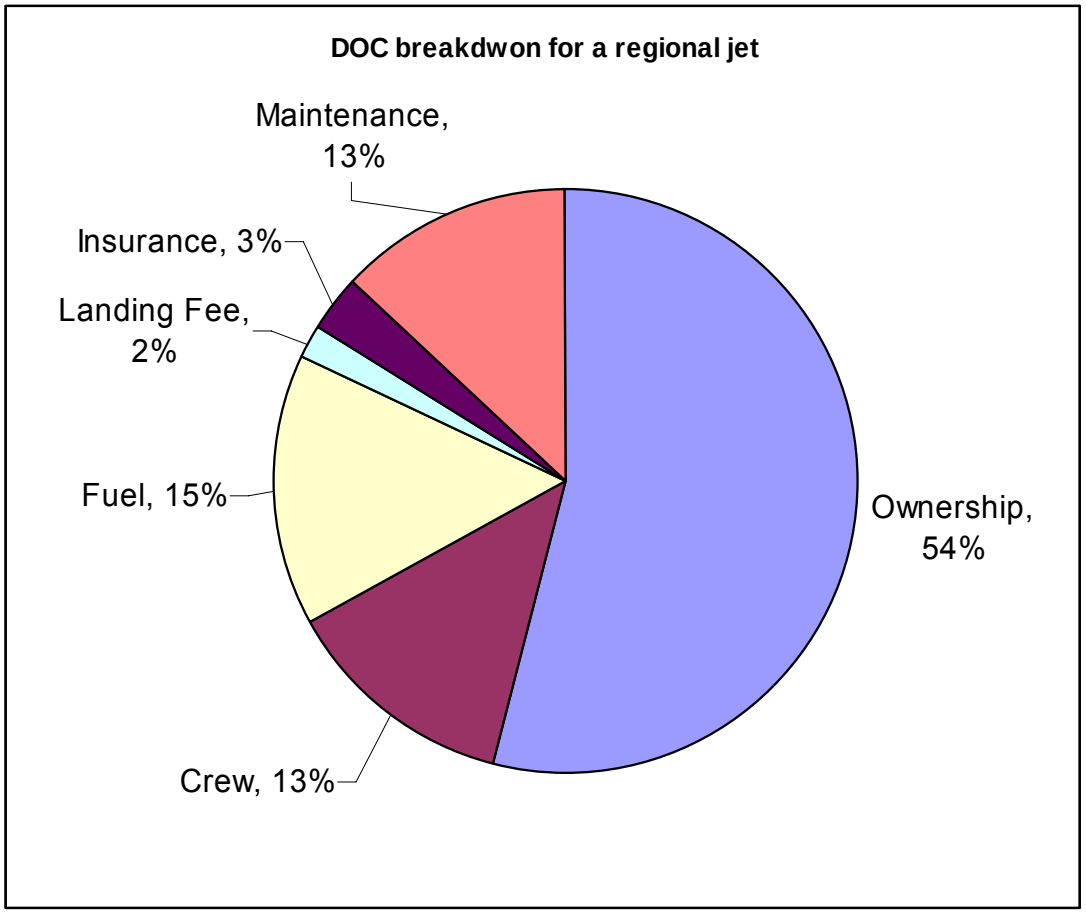

Figure 39 - DOC breakdown for a regional jet [49]

The fleet management part of the maintenance cost generates the overall cost to the airlines depending on the number of aircraft they use and the way they operate these aircraft. This is linked to the previous maintenance cost, and is in fact a representation of the cost for the overall fleet. Also it is important to see that the difference between short, medium, or long haul flight will have a great impact on the economics of the maintenance process. Indeed, short haul travel will have a greater effect on the deployment mechanisms due to the higher frequency of landing and take off. Also it is possible to analyse the overall maintenance cost over the predicted lifetime of the aircraft, this as well as giving the results for one aircraft. Depending on the flight pattern across their fleet, airlines will have different occurring cost for the maintenance of their devices, and also the more maintenance you need the more time the aircraft stays grounded and so there is a possible loss of profit for the airlines.

Figure 40 – Fleet management cost estimation [Lajux]

It can also generate the average cost of maintenance for the aircraft per kilometres flown per seat, this give a precise idea to the airline, or the aircraft manufacturer, of the cost of maintenance applied to the customers on their ticket depending on the distance travelled. This is a very important parameter as this cost will be transferred to the customer through the price of the ticket and so it will directly affect the airlines profit margin.

⇨ Mass, reliability and cost estimation conclusion

The tools developed can help to quickly assess different types of devices (or design option) at an early design stage. These tools have shown to provide clear and important answers to the user. The tools described in the preceding pages show that it is now possible to see the effect of different layouts for the LE in terms of mass and reliability. These last two parameters are in fact directly linked with the DOC of the aircraft. Less mass (for the aircraft structure) for the same aerodynamic performance, means more payloads being transported, and so more money for the airline. Also a more reliable LE system means a reduced maintenance cost. All of this directly contributes to the cost of running an aircraft, so it is important to keep these parameters to a minimum.

Different configurations can be tested at an early stage to show clear and rapid results.

This can prove to be very important for the aircraft manufacturer, who needs to provide a projected maintenance rate, average maintenance cost and DOC, as well the projected overall aircraft mass and maximum payload. These are basic data to be supplied to the airlines in order to market each type of aircraft, or to show the efficiency of different configurations. This way, airlines can decided on what is the best configuration (or aircraft) suiting their specific needs.

3.4. Aerodynamic design

The aerodynamic design for the wing concerns the design of the different elements of this wing compared to the aerodynamic efficiency and the given performance. From the type of LE device used, to the details about how to joint this LE device to the wing, there are a few steps to progress from the general design points to more precise details. This process includes a full fluid dynamic analysis and a description of different aerodynamic requirements.

3.4.1 Flap type

Different types of design have been described in chapter 2 and the parameters for the design of LE device are given in the preceding step of the methodology.

⇨ **General constraints**

- Front spar position
- Slat/flap shape optimisation
  - ⇨ depends on airfoil profile (on cruise, take off and landing condition)
  - ⇨ Span/Chord/Thickness
  - ⇨ LE radius
- Aerodynamic efficiency

⇨ **Constraints in kinematics design**

- Aerodynamic efficiency = Nb of slats/flaps
- Complexity of supports

The ideal solution is to compromise between the lowest amounts of slats elements with the least complex mechanism. The preceding paragraph provides the base for the choice of an optimum configuration.
Other solutions to further improve the aerodynamic characteristics are the use of Hybrid Laminar Flow Control (HLFC) or the use of smart materials able to change shape. However, the use of such technology will not be studied in the scope of this project.

3.4.2 Aerodynamic requirements

The aerodynamic requirements for any aircraft, also called performance requirements, are governed by official requirements listed by the Federal Aviation Administration (FAA) in the Federal Airworthiness Regulation (FAR). In the case of a large subsonic civil transport aircraft, Part 25 [38] of the regulation is applied. The detailed description of performance requirements relative to this type of aircraft can be found in sub-part B (flight – performance) of FAR 25.
For this methodology, it is necessary to consider the take-off and landing requirements. Any high-lift system, variable camber device or other flight control devices must comply with this set of operating rules, as any flight control will be used during these flight phases.

⇨ Take off requirements

For the Take off phase the FAR 25 specifies a specific climb angle (sin γ), also called climb rate (R/C) for ground roll, first segment and second segment climb. When analysing the performance of the aircraft during these phases, the most important parameter will be the climb rate, as it is function of the L/D and the Thrust to Weight ratio (T/W):

$$R/C = \sin\gamma = (T/W)\text{-}(L/D)^{-1}$$

Equation 3-8

R/C= Climb Rate [m/s]
T/W= Thrust to Weight ratio [N/kg]

Consequently the efficiency and deployment of the LE or Trailing Edge (TE) devices will have an impact on the L/D, and so on the climb rate.

During the take off phase, the take off field length is defined as the total ground roll distance to lift off plus the airborne distance to fly over a 35 feet (or 10.7m) high obstacle [10]. The lift-off speed (V_{LOF}) has to be equal or greater than 1.1 times (or 1.05 with one engine out) the minimum unstuck speed (V_{MU}). V_{MU} represents the minimum airspeed at which the aircraft can safely take-off with one engine operative[12] as V_{MU} is function of the maximum lift and the rotations. This causes problems as decreasing rotation will imply to increase the lift coefficient. But the angle of attack is restricted by the fuselage's tail-scrape angle, this can become a big problem when designing a derivative version of an aircraft with stretched fuselage.

Just after taking off the ground the aircraft must reach its take-off climb speed V_2 (before arriving at the screen height of 35 feet), which must be equal to or exceed 1.1 times the minimum control speed (V_{MC}) and must exceed 1.2 times the minimum dynamic stall speed in steady flight (V_{Smin}). V_{Smin} is usually about 0.95 times the stall speed in steady flight (V_{S1g}), so V_2 has to be greater than 1.14 times V_{S1g} [10]. This implies that the C_L must be equal to or less than $C_{Lmax}/1.13^2$ (where C_{Lmax} is maximum lift coefficient for a given flap configuration), which means that a high maximum lift coefficient is essential to obtain a low take off climb speed. This also means that it is possible to obtain a different C_{Lmax} for different LE & TE slat/flap configurations.

During the second segment climb phase, when landing gear is retracted, the aircraft must maintain a speed greater than V_2 (with one engine failed) and a R/C greater than 2.4% for a twin engine (2.7% for a tri-engine and 3.0% for a four-engine).

There is an obvious conflict between the short take off distance and the best climbing rate (with one engine out), therefore it is important to improve and optimise the aerodynamic efficiency of the wing and especially looking at the efficiency of the LE & TE devices.

⇨ Landing requirements

The landing phase can be divided into two steps, first the approach phase until touchdown, and the braked ground run phase.

The final approach is usually flown at a glide slope angle of 3°, with FAR 25 requiring that the approach speed must be at least 1.3 times V_{Smin}, which translate to about 1.24 times V_{S1g}. therefore the lift coefficient during approach ($C_{L\ appr}$) is about $C_{Lmax}/1.54$), and a low approach speed necessitate a high C_{Lmax}.

Most modern aircraft have no problem matching the landing with the take–off field length. The critical parameter is generally the approach speed. The higher the approach speed , with a low L/D, the more control and responsiveness the pilot has, but on the opposite side an approach speed too high will lengthen the landing rolling distance (for the same C_L)! The approach speed also has an effect on landing accident rate and economical consideration (tyres and brake wear). The final problem of the approach speed is that a low speed will necessitate a high maximum lift coefficient and that might reduce the pilot visibility and so the maximum aircraft's angle of attack during approach.

The R/C is also important if the aircraft has to abort landing, the minimum climb gradient is 3.2% (flaps deployed, all engine on and landing gear down), and as a result a high L/D is required. The second reason is that the visibility of the pilot will be reduced if the R/C is too high as the angle of attack of the aircraft will have to increase to match the required lift. The LE and TE are a mean of increasing lift to match this required lift with a limited speed. Deploying the LE and TE devices helps to increase the lift and also keeps the angle of incidence to a low value to help increase the visibility.

3.4.3 Geometrical constraints

The aircraft has to fulfil the different aerodynamic requirements, but there are also some constraints on the space available to design the LE device and its deployment system. These constraints are listed and described below.

The first parameter to know is the spar position (generally located 20-30% of the chord), as it will define the amount of space available on the front of the front spar. It will also provide support for fixing the deployment mechanism.

The wing can be subjected to excessive loading (twisting and bending) which means that the wing box will need to increase its structural stiffness. This is done by increasing the size of the structural elements (spars and ribs), which will reduce the available space for LE devices.

Another geometrical constraint to consider is any possible chord wise extension of the LE; this will have an effect on the length of deployment.

All the precedent elements are chord wise constraints, but there are also constrains to be considered span wise. First is the position of the engine and the engine pylons (or nacelles) as it will dictate the spanwise location of the LE panels. It is unlikely that the LE panels will be deployed on top of the engine(s).
Another spanwise constraint will be the configuration of the wing tip (or winglets if there are some), but this will be analysed in a later paragraph.

The maximum thickness of the LE device will be defined by the thickness of the wing at cruise condition.
These parameters will "feed" the structural mechanism design part of the methodology for further design considerations regarding the kinematics and structural integrity of the system.

The available space to package the LE mechanism is critical at this stage as the best way to design the LE will be to reduce the cost and weight of the mechanism, which is done by using a mechanism as simple as possible

3.4.4 Initial flap airfoil

The initial flap airfoil can be designed, as the geometry is now fully known; all the geometrical and functional constraints have been previously described.

It is then possible to generate the profile of the initial flap, this includes the position of the LE and TE spar (compare to the overall chord), and also any possible chordwise extension. It is also possible to generate the different camber positions for the airfoil.

A specific Excel spreadsheet to generate the different camber for an airfoil, including LE and TE camber variation has been developed by the author (Figure 41 in page 63). This new tool quickly generates new sets of coordinates for the new airfoil, with the different cambers specified by the user. This tool also gives the user opportunity to change the centre of rotation of the LE and TE deflection.

From this point it is possible to select an airfoil which has performances close to "the design point". It is also possible to refine this airfoil profile and characteristic by reducing or eliminating some of the drag or improving lift. This can be done by using some specific parameters of the airfoil profile and to see their effect on the overall airfoil performance [2]. Computational Fluid Dynamics (CFD) tools are normally used to analyse the performance of the initial airfoil. This is used as a design tool before developing models for wind tunnel testing. It is both a cheaper and quicker iterative process than testing a model on a wind tunnel. This is done as an iterative process to further improve the characteristic of the initial flap airfoil. This simulation process will be described in more detail in the next paragraphs.

This part of the methodology gives the generic aerodynamic shape of the airfoil as the envelope and the available space to fit the LE device. A detailed set of points (coordinates) is generated to represent the airfoil; this data will be used at a later stage for the mechanical design.

The tool developed for this application provides users with a rapid way of generating 2D profiles (and associated coordinates for data) within a short amount of time. It helps to create fully cambered wing profiles.

EDI MIDDLE SECTION:

x	y
1.000000E+01	0.000000E+00
9.903930E+00	1.384000E-02
9.619400E+00	6.190000E-02
9.157350E+00	1.362200E-01
8.535530E+00	2.248800E-01
7.777850E+00	3.184500E-01
6.913420E+00	4.044500E-01
5.975450E+00	4.713700E-01
5.000000E+00	5.121900E-01
4.024550E+00	5.256000E-01
3.086580E+00	5.130700E-01
2.222150E+00	4.794700E-01
1.464470E+00	4.258500E-01
8.426500E-01	3.504900E-01
3.806000E-01	2.537600E-01
9.607000E-02	1.358000E-01
0.000000E+00	0.000000E+00
6.070000E-02	-1.373700E-01
3.806000E-01	-2.601200E-01
8.426500E-01	-3.648400E-01
1.464470E+00	-4.510600E-01
2.222150E+00	-5.159800E-01
3.086580E+00	-5.517400E-01
4.024550E+00	-5.541900E-01
5.000000E+00	-5.240000E-01
5.975450E+00	-4.565700E-01
6.913420E+00	-3.456100E-01
7.777850E+00	-2.149700E-01
8.535530E+00	-1.121700E-01
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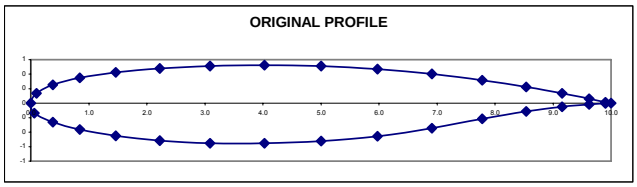
9.619400E+00	-1.764000E-02
9.903930E+00	-2.100000E-03



Modified profile=

Leading edge centre of rotation= X= 2.00 Y= 0.00

Camber var.= 15.00 deg 0.261799 rad

Trailing edge centre of rotation= X= 8.00 Y= 0.00

Camber var.= 0.00 deg 0 rad

Figure 41 – Variable Camber profile creation using Excel spreadsheet [Lajux]

3.4.5 Layout/configuration

Once the initial profile has been created the next step of the methodology is to look at the overall configuration of the LE device installations along the wing.

The first element to look at is the span, and how much length of this wing is available to position the LE devices. The longer the wing span is, the more effective the LE devices will be. However when considering the span of the wing it is also necessary to rely on what is on the wing and how the wing is arranged.
Often there are one or two engines on each wing and with each one of them there is a nacelle (or engine pylon) which provides the structural link between the engines and the wings. The LE device will have to be split into a few elements (or sections) along the span as there should be no LE devices where the nacelle is connected to the wing. Generally the LE devices are not deployed on top of the engines.

At this point it is important to consider the case of when the LE is deployed. The angle and the range of deployment should be carefully considered to avoid potential clashes with the engine during deployment.

However it is best to keep the spanwise aerodynamic continuity of the LE as this will have a strong impact on lift and drag. A possible solution to reduce the impact of the

spanwise discontinuity, at the nacelle position, is to use engine nacelle chines [10]. Once the wing separation (LE device compare to engines position) is achieved it is necessary to investigate the configuration of the wing root and wing tip. Depending on what design is chosen for each of them, then the available length of span for the LE device will be reduced.

The configuration, and shape, of the wing tip might reduce the possible outboard section of the LE. For example, if winglets are used that will decrease the available space for the outboard section.

The same kind of problem applies to the in board section as the wing root configuration might reduce the available space to fit the LE device. For example when the main landing gear is attached at the wing root, then its stowed position will be within the wing root part of the wing. This could interfere with the LE device space. All of these different constraints will create “gaps” (and discontinuity) on the LE of the wing. These “gaps”, without LE device deploying, will be located within the different LE sections and also at the wing tip.

These “gaps” on the span cause a loss of possible lift proportional to the loss in LE area and also the creation of turbulence (vortices) on the sides of each LE device.

Once this is all known it is necessary to investigate the design possibilities for the interaction at the extremities of each LE devices. One classical solution is the use of closing LE ribs (or riblets) which fill the gap at the extremity of the LE and have the shape of the general airfoil.

The exact length available to put the LE devices is defined, so the next step is to look at the split of the LE device into several sections. Each one of these sections is made of several panels. This is generally done by having a set of inboard LE and another set on the outboard of the wing. However, for larger aircraft (such as the A380 or Boeing 747) there is the possibility of having a central section. This occurs when there are two engines on each wing. This will be very important at a later stage, when doing the structural analysis and the kinematics design. It will be possible to establish exactly how many elements (or panels) will be on both the inboard and outboard of the wing. This will depend on the resistance and bending characteristics of each panel of the LE.

Figure 42 shows the example of selected wing layout from the Airbus A320; this shows exactly where all the different main components of the wing are located.

Another thing to consider is the possibility of LE chordwise extension since this will have an influence, and will interact, with the other components on the wing. The extension and deployment motion has to be evaluated to make sure that there is no clash between the different components.

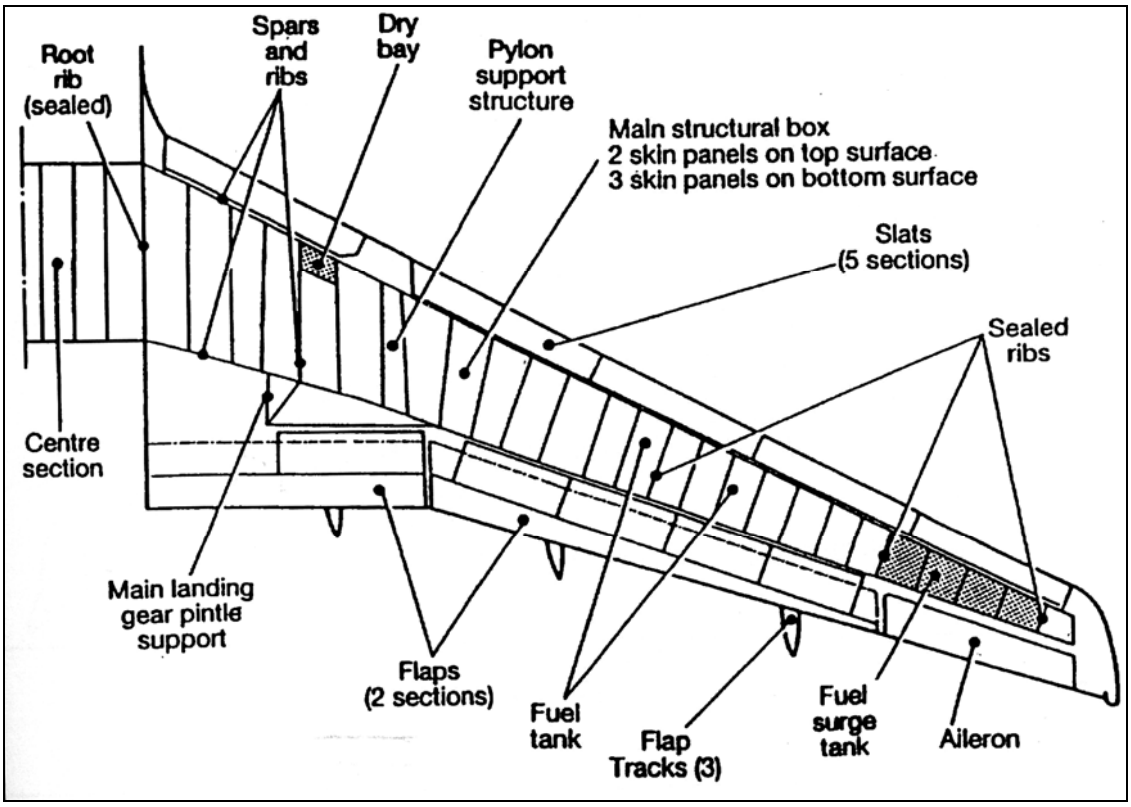


Figure 42 - Example of wing layout (A320 wing) [10]

The final point to consider is the potential protrusion of elements outside of the airfoil. It would be possible to imagine that the LE deployment mechanism might be located outside of the airfoil. In this case it would be necessary to add a fairing that would have an obvious impact on the aerodynamic performance of the aircraft.
All these points need to be considered when selecting the overall wing layout as they will have a major influence on the aircraft performance. These performances will be analysed in the next paragraph.

3.4.6 Check of aerodynamic performance

The aerodynamic performances of the designed airfoils have to be checked to be sure of the efficiency of the wing compared to what is expected of it. As the wing is totally designed, including all the elements, it is necessary to look at the exact aerodynamic behaviour as well as the loading applied to the wing. This is be done using different computer programs at first, and then to do some physical testing before going through test flight phase.

- ⇨ **Aerodynamic results**

Using CFD tools combined with the data from the preceding paragraphs enables users to study in detail the aerodynamic performance of the chosen wing or airfoil and also to see the effect of variation of the LE position.
The analysis should be carried out on a 2D case to start with, which can be later expanded to a full 3D model. However, doing a 3D analysis requires a lot more time to get the results due to the amount of computation time. It is easier to do a 2D study of the airfoil and then use Engineering Sciences Data Unit (ESDU) datasheets to estimate the aerodynamic performance of the full wing.

For the 2D study it is necessary to design the airfoil profile to be used for running the simulation. This one could also be modified at a later stage to create the LE changes to represent the variable camber modifications.

- ⇨ **Low speed study (landing and take off analysis => M=0.2/0.3)**

It is necessary to study the aerodynamics of the aircraft at low speed for M<0.3. This type of speed represents the usual speed for take off & landing. It also has to be noted that this low speed case does not have supersonic speed, or sonic boom across the chord of the airfoil. So the wing can be tested using a CFD software for low speed (subsonic) and inviscid flows.
For the study of the low speed condition, it is possible to use the Xfoil software to quickly do the analysis for the selected airfoil. This software provides quick results for simulation due to the viscosity of the air, or for inviscid flow for M<0.3. It uses a panel method[39] to describe the airfoil for the analysis. This method is quick and simple to mesh the airfoil.
Xfoil also provides the Cp distribution along the airfoil for different angles of attack. Using this software it is also possible to find the C_l or C_d and get the L/D graph. It is also possible to fully define the speed at which the airflow moves around the airfoil.

- ⇨ **High speed (cruise => M>0.6)**

The study of the high speed case, for speed above M=0.6 generally implies that the flow might become supersonic and that the viscosity of the air must be considered. The viscosity will cause further drag and so will increase the overall drag, and then increase fuel burn. It is also important to consider the possible creation of shockwaves on the airfoil as the air speed can become supersonic.
For all these reasons it is important to use software which will able the designers to represent the real case as accurately as possible. So for the high speed case it has been

decided to use the Fluent CFD code, as it provides a more accurate tool to analyse the airfoil performance at different speeds. This program is also very efficient and accurate; it is possible to define the flow conditions, the turbulence model, etc… However, using Fluent also means that the user will have to create the airfoil profile and the matching mesh for this airfoil. On top of this the user will have to fully define the boundary conditions (for the "box" around the airfoil) and the flow and pressures going in and out of this box.

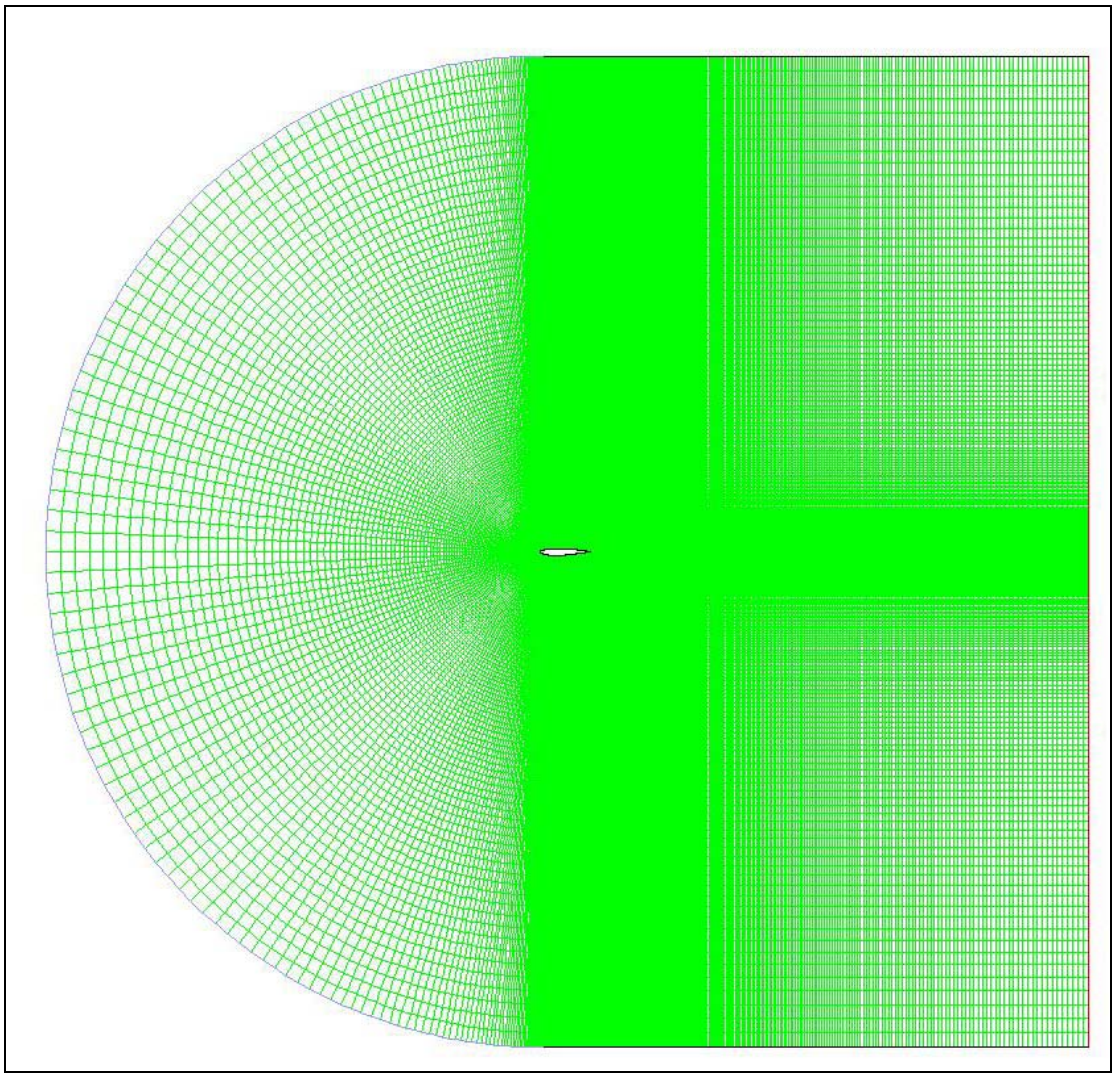


Figure 43 - Example of CFD grid [Lajux]

Once the airfoil profile has been generated and fully defined using the Gambit meshing software it is possible to start making the mesh for the CFD study. This mesh will have to be coherent to the kind of study carried out. This mesh will have to be more refined at the LE (see Figure 43 for example)and around (including upstream) as this will be the main area of interest for this research, but it is also important to keep a refined mesh along the airfoil skin to identify the boundary layer. It is also important to have a refined mesh downstream of the airfoil to catch the wake and to represent any turbulence and get a

more accurate result for the drag. All of this can be done using meshing software (such as Gambit or Gridgen); this software allows the user to make the grid for the CFD study. The meshing software is also used to define the type of boundary conditions for the model as they will be used during the CFD simulation.

Once the mesh is done, the user can input the data related to the case conditions including the boundary conditions and the flow characteristic.

After the mesh, all of the data (mesh + profile + boundaries) can be used with a CFD package to run the simulation. At this stage it is necessary to input physical parameters to represent the case:

- fluid description
- viscous model
- Energy equation / model
- Turbulence model
- Reynolds Number
- Speed

Once all the different parameters have been carefully described it is then possible to run the simulation. This run could take up to 2 or 3 days depending on the complexity of the model. Also the computational power of the machine use has to be considered, in fact by using supercomputers it is possible to run much larger simulation within less time. However, the cost of purchasing and maintaining such a facility is quite high. This is why a compromise has to be found between the power of the computer and the time to generate the results.

At the end of the simulation the user can collect a set of data representing the performance and behaviour of the wing (see Figure 44 in page 69), or airfoil, in the given condition. It is also generally possible to see if there is any shockwaves happening along the airfoil profile.
It is possible to get different data as output in order to analyse the performance of the airfoil or the quality of the solution.

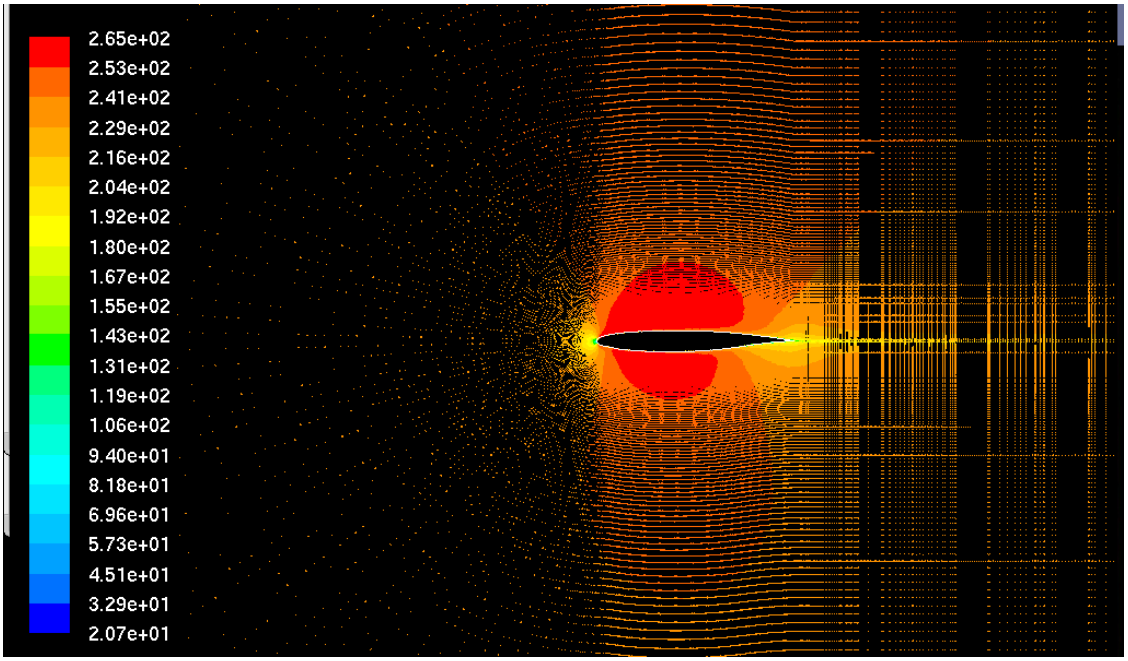


Figure 44 - Example of CFD results using Fluent [Lajux]

⇨ Loading results

Using CFD tools it is possible to generate the pressure distribution for an airfoil at different conditions (speed, angle of attack, etc…). CFD programs such as Fluent or Xfoil can generate the pressure distribution around the airfoil. This is represented as a graph showing the C_p at different chord wise positions. Generally the upper line represents the C_p applied on the airfoil upper surface skin, and the lower line represents the C_p applied on the airfoil lower surface skin (see Figure 45 for example in page 70).

From this set of data it is possible to use a spreadsheet to process the data and to calculate the discreet chord wise loads distribution. The passage from the pressure to load is done using the area under the Cp curves on the pressure distribution or using CFD tools to directly generate C_l.

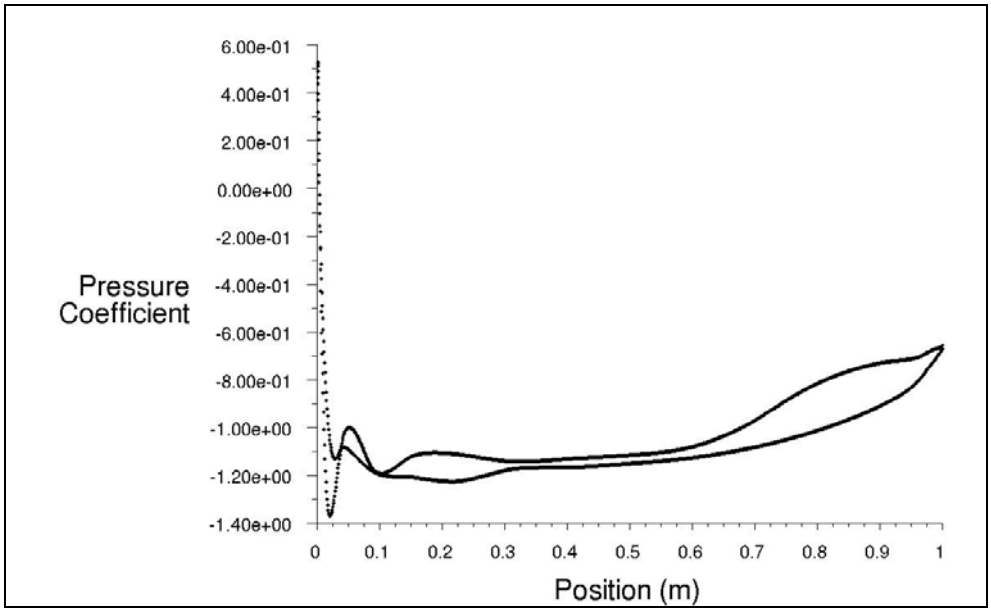


Figure 45 - Example of Cp graph [Lajux]

The pressure distribution on the wing can be calculated using the basic formula given below:

$$P = \frac{1}{2}\rho V^2 c C_P$$

Equation 3-9

Where:

P=	Pressure	[N/m]
C_P=	Pressure coefficient	[--]
c=	Chordwise of local wing section	[%]
ρ=	Density of air	[kg/m^3]
V=	Aircraft speed at specific flight conditions	[m/s]

Using the results generated with the CFD simulation it is expected that users can generate a full set of data to represent on a graphical form the pressure gradients both on the chordwise and spanwise direction. This way it is possible to analyse the behaviour of the wing and see where the high load (or high pressure zone) will occur.

After getting a set of results from this aerodynamic study, the user can develop new sets of airfoil and generate modified airfoils to see if it possible to improve the performances based on the initial profile. This is a manual optimisation process, which can take a long time to perform due to the large amount of parameters available; this includes airfoil shape variation, speed changes, etc…

Also following on the CFD simulation it is normally expected to run wind tunnel testing to validate further the results obtained during the simulation phase, this can be followed by flight test to finalise the overall aerodynamic characteristic of the designed wing.

3.5. Structure & mechanical design

The design of the LE devices deployment mechanism is generally a complex problem due to the lack of space available to package it. Also the lack of potential "hard points" to hold the mechanism to the wing makes it even more complex. The design solution will also have to comply with the complex deployment trajectory. The other difficult task is to be able to define a design solution which is "structurally" correct, which will not break under the loading occurring during the different flight phases.

3.5.1 Kinematics design

The kinematics design part of the methodology will investigate the different possibilities and kinematics solutions available to fulfil the objective set from the aerodynamic results. The mechanism designed for the deployment of the LE device must follow as accurately as possible the deployment trajectory. It is also very important to consider the surrounding components (engine, nacelle, landing gears…) to avoid any clash between them (as mentioned in the precedent paragraph on layout design).
This kinematics design study will be carried out by doing a first design. This design will be using only a 2D model of the airfoil and deployment trajectory. This is to find out the general form of the kinematic chain for the mechanism. Once the 2D model of the mechanism is designed it is possible to investigate the problems linked with a 3D development from the 2D solution.

⇨ 2 D deployment mechanism

To start the design of a new LE device it is necessary to get the exact data from the aerodynamic initial study, which provides the following details:

- a 2D profile of a generic airfoil
- the position of spars
- the position of fuel tanks (if there are fuel tanks in the wing)
- the wing thickness

The deployment trajectory….

Once all these parameters are known it is possible to use CAD tools to generate a simple design incorporating all of this. This CAD file is then used to develop the first kinematics chain which will comply with all the geometrical constraints (i.e. available space).
The research of possible solutions can de done using a "trial and error" method, or by using new CAE tools such as the SYNAMEC software. Using this type of software helps to create 2D mechanism. This program will generate a 2D (X/Y) stick diagram including kinematic jointts [40]. For this research it has been decided to use new design product such as the SYNAMEC software to investigate innovative solution and also to save time compared to the "trial and error method" which can take a long time and not necessarily give good results.
The SYNAMEC software, which was developed during a European Union research project[29], provides kinematic chain to the user using fixing points and deployment trajectories. This tool is used for the automatic generation and synthesis of new type of mechanisms. It generates kinematic chains but it also provides the user with

representation of the kinematics jointts between the different components. The automatic generation of mechanism does save a noticeable amount of time to the user.
The 2D kinematic chain for the LE device deployment is created, but it might not be fully optimised to start with. Some of the coordinates (of the points) should be used as parameters to optimise the quality of the deployment trajectory.
By giving a range of values to the support points (or parameters), linked to the front spar or to the mechanism dimension it is possible to improve the result. This improvement can be done using optimisation software link to the CAD system to accelerate the design process. By investigating the possible change of geometry depending on the range of variation of the parameters, it enables engineers to design a more accurate 2D mechanism.
This same process can then be applied to another section of LE device along the wing to "scale" the initial mechanism and obtain an optimised geometry.

- **3D problems**

This addition of 2D profiles, of the different size along the span, is the "skeleton" for the full 3D mechanism. It is the basis on which the full mechanism system will be designed.
Having designed 2D kinematics for each section, it is possible to investigate the potential problems of the mechanism becoming a 3D system. As it becomes 3 dimensional, there are more problems to solve; they are linked with the deployment of the different mechanisms geometry moving at the same time.

The first thing which needs to be considered is if there are any clashes between the LE devices and other part of the aircraft such as the engine, the landing gear and other elements. During the deployment of the LE panels it is possible that they will clash with the surrounding components. This type of problem can occur in both the chordwise and spanwise direction.

The second point to analyse is related to the problems occurring when deploying a 3D mechanism along the wing. Since the wings have a sweep angle and the LE device also has a deflection angle this means that the all deployment motion must be conical. That generates many kinematics problems as the geometry of the LE mechanism must be of different dimensions from the root to the tip of the wing. Each mechanism has to be "scaled" along the wing.

This conical motion also generates potential jamming problems during deployment. The complexity of this motion can cause the overall mechanism to jam as some of the components can get stuck if the manufacturing tolerances have not been respected or if it has been used in extreme conditions.

However, if cylindrical motion is used for the LE device deployment it is certain that there will be less time spent on the design and optimisation since the same geometry will be reproduced along the wing. If the same mechanism is not reproduced along the whole wing at least each section (inboard / central / outboard) will keep the same geometry for the LE mechanism.

At this stage, the results obtained are normally a simple stick diagram (including kinematic jointts) for the overall LE deployment mechanism. This represents the 3D layout of the deployment system including the different parts of the mechanism along the span, which is referring to the number of LE panels on the wing

3.5.2 Structural design

The structural design part of the methodology is dedicated to the initial study and a more detailed study of the structural characteristic of the mechanism and associated components including the LE panels. There will be also a description of the loads applied to the LE devices (and its mechanism) during deployment, and during the different flight phases.

⇨ Initial structural design (slender beam analysis)

It is possible to "extract" the exact geometry of the LE mechanism from the kinematics study. This geometry is used to start the Final Element Analysis (FEA) study of the mechanism.

The first step is to use this data and to make a structural model, using a simple 2D analysis. The first "structural" model is done using slender beams to represent the major components of the mechanism. A 2D study will take less time then a full 3D detailed study, and will give the general results for the components geometry.

Each beam is defined in term of section geometry and length, the length being taken from the kinematics study and the section is defined by the user. That might have to be optimised at a later stage to improve the weight saving.

Each beam material properties are defined based on experience and materials normally used for these kinds of mechanism. These parameters might be later optimised to lower the cost and weight of the LE mechanism

These beams are then meshed (using a mesh generator) to be prepared for the rest of the analysis. This step is very important since the definition of the meshing will influence the accuracy of the final result.

Once this first structural model is ready it is possible to apply the loads. These loads come from the aerodynamic part of the methodology; the loads are calculated through the pressure applied to the LE device. The total pressure divided by the LE area gives the load applied to the LE device, this can then be divided by the number of LE panels, and then divided again by the number of supports for the LE mechanism into each part of the mechanism.
At this stage, it is recommended to use a simple set of loads to approximate the section dimensions and the material to choose in order to design a system which will resist these sets of loads.

The results obtained from this first analysis can be used to either transfer to a more detailed 3D analysis or can be used to further improve the 2D analysis and run a simple optimisation.

⇨ **Detailed structural analysis**

Using the results and geometry defined in the 2D structural analysis it is possible to go to a 3D study. This will give a more detailed result and should provide finally a result with a fully defined 3D set of components.

Mechanism components detailed structural analysis:
The first step of the detailed structural analysis is to define each major elements of the mechanism as 3D elements with all the exact dimensions.

Once each element has been modelled it is then possible to create the 3D mesh (using a mesh generator), that mesh normally requires a lot more time to generate than for the 2D study. This mesh will have to be refined in the area of high stress to be sure to have a detailed evaluation of the amount of stresses going through each component.

The materials properties can then be added on to the model.

Finally, the full loading can be applied to the 3D mesh. It might be possible to run simulation under different sets of loads to represent different situation such as landing, cruise or take off. That would help to understand the structural behaviour of the chosen design under these sets of loads.

At this stage of the FEA 3D study it is possible to collect the first results and it is possible to analyse them and to improve from the first solution. It is likely that a manual or automatic optimisation will be run. This optimisation will further reduce the weight of the different components.

The same process can be used for the design of the slat structure, including skin and stiffeners (stringers, ribs, spar). This process can be used specially to analyse the panel maximum deflection under critical or ultimate loads.

3.6. Integration of aerodynamics and structure

To finish the design phase it is important to link the results obtained during the aerodynamic phase and the mechanical/structural design phase.

The aerodynamic design phase gives the planform and general geometry of the wing, as well as the wing layout, and airfoil profile.

On the other side the mechanical and structural design phase gives the geometry and size of the deployment mechanism, and the envelope to fit the LE.

The mass and reliability estimation tool also provides important data related to the final mass of the overall LE mechanisms as well as the overall reliability of the system. The

reliability data are very important for the airlines using the designed solution. It has a direct impact on the DOC. Overall associated cost is estimated through the cost estimation module to further refine the solution and show the financial cost of each type of devices and configurations.

The CFD simulation & mechanism design (including FEA) helps to develop a more accurate solution and also more effective. After going through these steps of the methodology it is possible to generate a virtual model for the LE device design solution.

At this stage it is possible to develop a prototype for final testing and to do further refinements of the LE device. These refinements can be done after some ground tests or flight tests or using a scaled model for the wind tunnel testing. They are finishing touches to the overall design, but there are generally no major modifications overall.

3.7. Detailed assessment of mass, reliability and cost

Once everything has been analysed, it is possible to refine the parameters of the mass and reliability to obtain a more accurate result. By using data from the virtual model the user can get the mass of the different component (by deriving the mass through density for different materials). Also by having all the components it is easier to count the exact number of parts, and to link this to the reliability estimation to get the final overall & detailed failure rate. This can also be used for the calculation of a more accurate cost figure.

3.8. Methodology conclusion

This chapter has shown that a new design process can be used for the design of conventional LE devices and VC LE devices. The methodology has been described for each step in accurate detail, especially when new tools have been developed for specific applications. This new methodology provides a new and innovative way to design LE devices including VC concept, this not only provides a way to design VC LE devices but also include totally new tools allowing the user to investigate and estimates the effects of different concepts at an early stage of the design process which is a completely new approach compared to other methodologies described in Chapter 2. The methodology created by the author has the advantage to focus on the design of the LE devices including VC concepts, which was not covered by the most common high lift methodologies [6, 55] the consideration of using VC LE at the high speed was not covered in the in the methodologies by past author. Also no integrated tools for assessment of mass, cost and reliability both for classical and VC LE devices has been done before and applied to a full LE device design methodology.

The complete design process has been explained, and it is clear that the use of new tools during the design process will surely reduce the time to a better design and refined prototype. And these tools will also enable aircrafts designers to get a better understanding of the effect of the different types of LE devices at an earlier design stage of the design process and to design an optimised solution at the first attempt.

This methodology has also proven that it is possible to imagine a new way of thinking when designing LE devices. They don't have to be designed when everything else has been designed and the LE device has only a reduced space to be fitted such as Flaig and Hillbig methodology [55]. This increases the time to design due to the complexity of the possible solution. LE devices can be considered at early stage in the design process and this enable engineers to see the potential effect of using different types of devices (or different layouts) for the overall LE system. As explained earlier in this chapter the choice of the LE devices can have a very important impact on the DOC and the maintenance cost for airlines.

This methodology has also proven to be really useful for the design of VC LE devices. This has made the design process easier for VC applications as they usually have more complex mechanisms and also designers are reluctant to use them as the mass and reliability were thought to be generally worse than other more classical solutions. This methodology also provides designers with a quicker solution and faster design process for such complex applications which is more efficient than the other methodologies (ref to chapter 2) as it includes new and innovative specialised estimation tools.

This is why this methodology, if used to its full extent, will reduce the overall cost of the design process and the manufacturing, maintenance, and cost, but will also generate more innovative solutions.

It has to be noticed that the design process explained in details in this chapter concentrated on the design of LE devices. However, the methodology diagram (see Figure 34 in page 41) has shown that this design process can be integrated within the overall wing design to provide a more optimised wing design solution. In fact, the effect of different LE devices types and configurations especially when VC LE devices are considered can have a great influence on the wing design. Research from Zink and Mavris [56,57,58] and also Pendleton [59] have shown that the control laws and the wing design can be dramatically changed using VC concepts, thus changing the wing loading , aerodynamics performances of the airfoil and inertia characteristics of the wing. Similar effects related to the use of VC LE devices could be shown using the feedback loop to the MDO wing design shown in the methodology, but this is out of the scope of this research.

As mentioned in this chapter's introduction, for the purpose of this research only the part related to the LE device design have been studied in more details.

This methodology will be used and put into practice at a later stage in this document (see chapter 5) with a case study and a set of results showing the full process and improvements provided when using this new methodology.

Chapter 4

4. Validation of the Methodology

4.1. Introduction

For any analysis or simulation it is necessary to establish a way to validate the results obtained either against physical testing results or against results of validated analysis using the same tools. For this case study no physical testing or experimental work had been carried out, as it will be too expensive to manufacture a full wing prototype. So the analyses were only based on specific computer software. This chapter describes briefly why each software had been chosen as well as example of validation for the results for the different part of the methodology, where analysis has been done. There is a detailed explanation of how the obtained results compared to already validated results both for the aerodynamic study and the structural study.

4.2. Validation of aerodynamic analysis

For the validation process of the aerodynamic analysis the FLUENT software has been used as it was the best commercial tool available at the University.

Fluent has been chosen to carry out the second stage of the aerodynamic analysis as it can produce a higher accuracy of results as well as being able to simulate turbulence and transonic conditions. This software is also very useful to generate the data to produce the loading for the structural analysis.

When using CFD it is necessary to understand the effect of the grid shape compared to the results quality and computing time. For this reason any 2D analysis in this report a structured mesh will be used as it is generally more efficient than an unstructured one. Structured meshes (or grids) provide a result with fewer errors as well as improving the time to compute the result.A structured mesh requires more time to set up from a user point of view, but this adds to the quality of the final result. All grids for this research have been done using Gambitt, the grid generation software which is linked with Fluent.

A necessary thing to show when doing a CFD study is that the results obtained do not depend on the mesh, this is called the grid independency. It shows that further grid refinement will not improve the final result.

The best way to show the efficiency of this scheme is to compare a few grids for a similar test case. Each grid will be more and more refined. The result of the CFD simulation is then compared against the difference in number of cells. A grid (or mesh) with more cells takes more time to compute, but does not necessary gives a big improvement on the result. Any grid improvements would only generate results which are slightly more accurate, but would require much longer computation time.

To prove that the results obtained for the case study are valid it was decided to study an airfoil and to run four cases (with a different grid for each case). This airfoil will be similar to the one used fir the case study. Then each case will generate a set of results to be compared to the other cases. This will give the best grid, compromise of computing

time and accuracy, which will be used for the case study at a later stage. This is the ideal method to test different mesh variation of a same profile.

The result of this validation are displayed on different graphs, first the number of cells for each case and secondly the Cp distribution and the static pressure). It is clear that case 4 use nearly twice as many elements than case 3 (Figure 46), but the pressure distribution is nearly the same (Figure 47). Figure 48 shows the similar pattern for the static pressure difference, but as there is little changes in the static pressure at the boundary layer than a refined mesh will only show a small improvement in quality of the result.

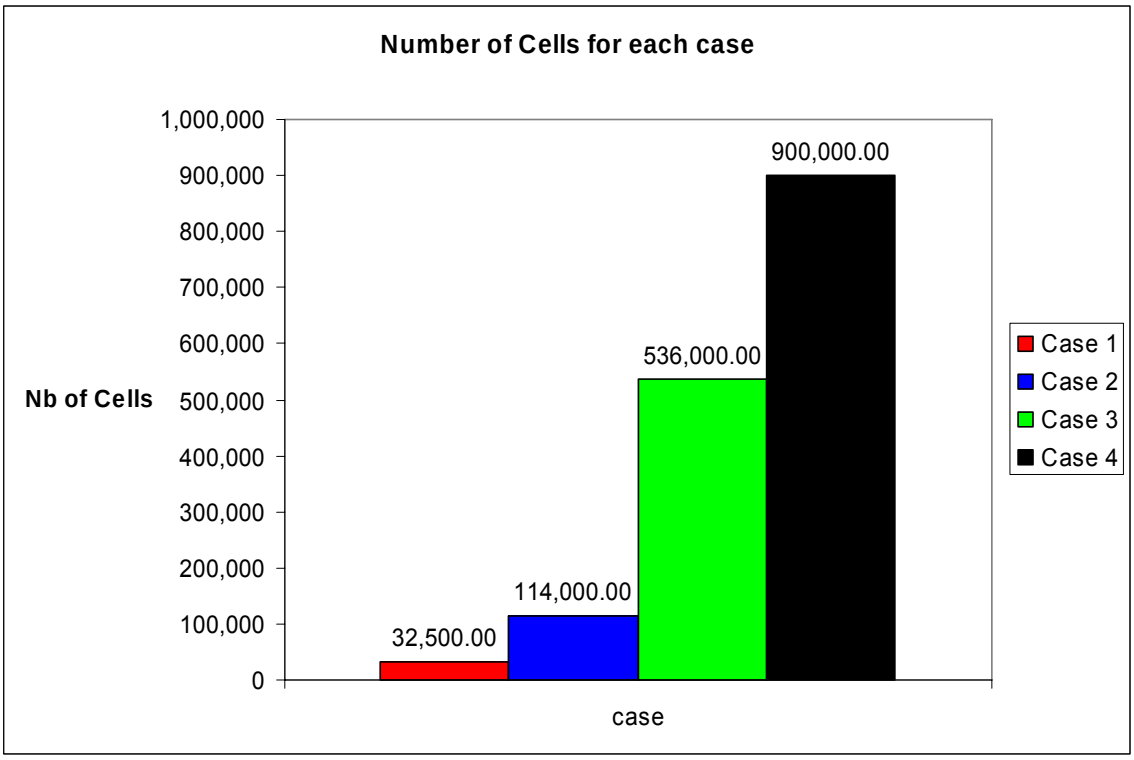


Figure 46 - Mesh quality difference [Lajux]

Following the grid independency it was decided that the setting for the mesh of case 3 will be used for the case study in the next chapter.

Another point to raise, when looking at the validity of the results, is to see the convergence of the different residuals as well as the convergence of lift and drag.

Finally it is necessary to keep the same parameters (solver, boundary layer…) as the one used in case 1 to case 4 in order to keep the same conditions for the CFD simulation.

By doing all of this validation process, the results achieved for the CFD analysis will be valid and reliable as the mesh has been checked.

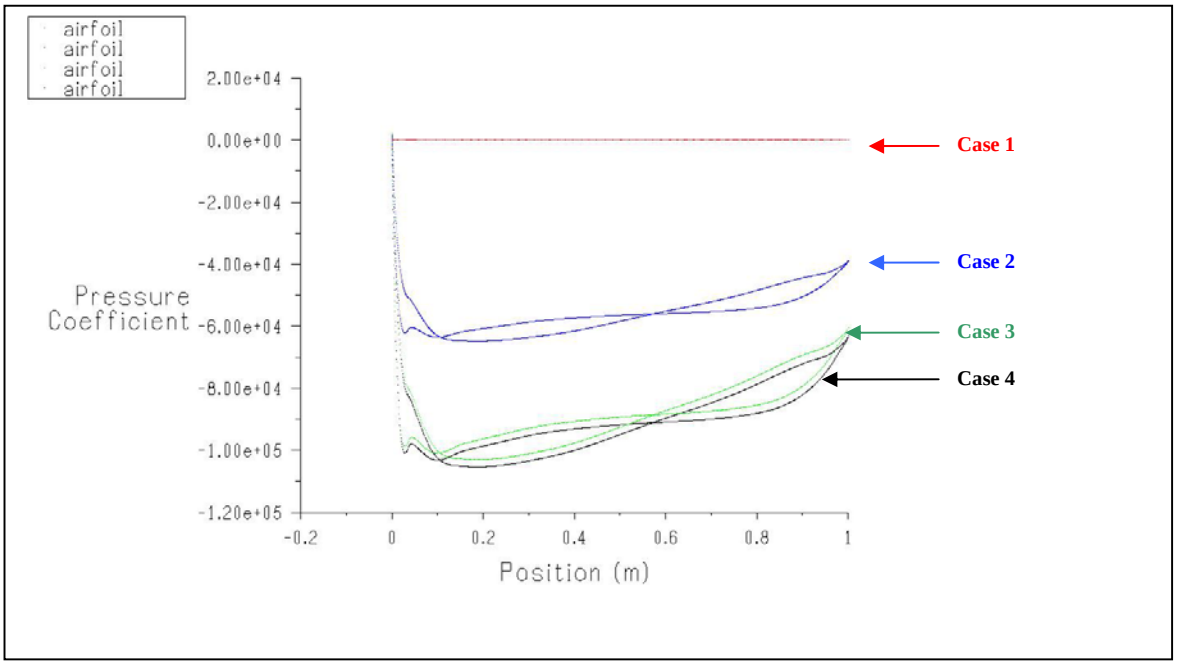


Figure 47 - Pressure coefficient difference [Lajux]

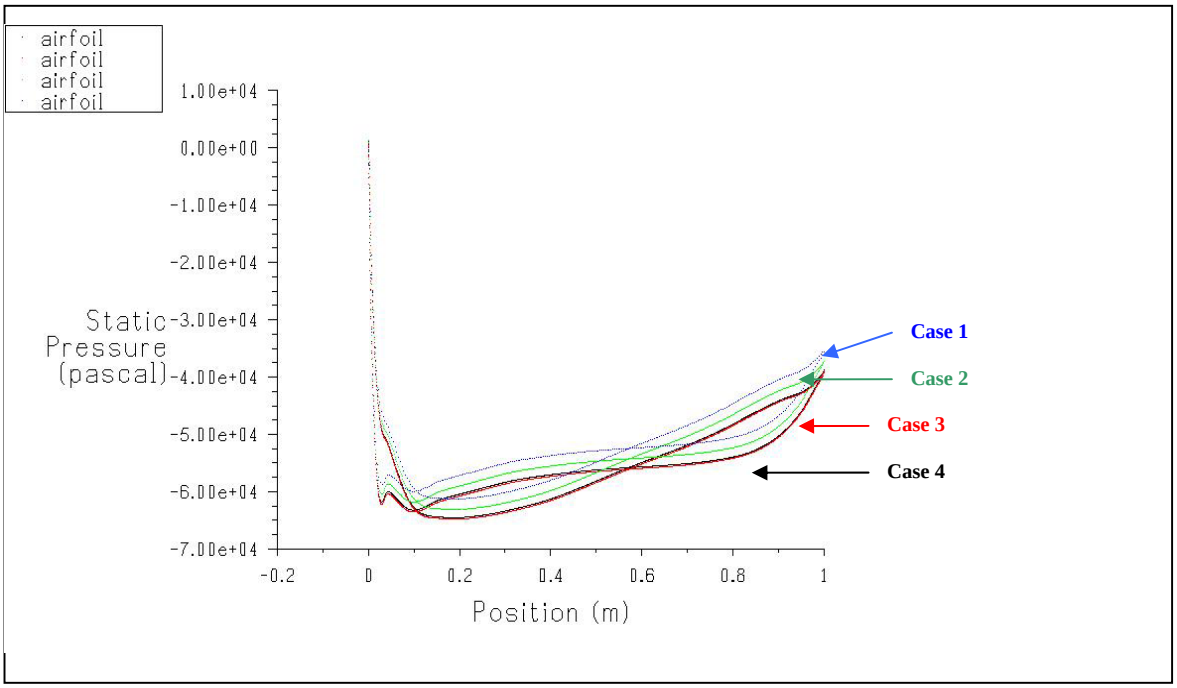


Figure 48 - Static pressure difference [Lajux]

4.3. Validation of structural analysis

For the validation of the structural analysis, it is necessary to prove that the CAD models and meshes are correct and that they will provide a valid solution. By keeping the mesh clear, well structured and refining it sufficiently, the user should get a correct set of results. This type of study will be used to validate the mesh used at a later stage in this report as applied to the structural study of the skin and the structure of the LE panel in the case study.

The following will demonstrate how it is possible to establish the ideal mesh for a full LE panel structural study. This includes a description of the meshing process by doing a comparison between different meshes for the same CAD model. Starting with a very simple mesh, each mesh will be increasingly refined. The more refined the mesh increases the computing time it will take to get a solution, however this means the solution provided, for maximum displacement and stress, will be more accurate.

By using the correct mesh, the user will be able to set up different set of loads to see how the structure behaves. These loads will represent the different loads applied during the different flight phases.

In order to see when the mesh is sufficiently refined it is necessary to find out how the deformation, or maximum stress in the structure, changes depending on the mesh density and quality. In fact, the stress and deformation should converge toward a specific value. There should be a point when the increased number of nodes will not massively improve the solution (within 5%-10% difference). Generally it is accepted that a small improvement on the solution at that point is not necessary as the computation time to reach the solution is much longer. At this point, the time spent on improving the accuracy of the result is not worth the amount of time wasted in doing it. A variation under 10% can be acceptable, but a 5% variation would be recommended. Any variation under 1% would be very accurate.

It was decided to use the CAD model for a LE slat component, including the skin and substructure, in order to validate the model which will be used at a later stage. This model will not only validate the results which will be obtained when doing the test case study, but it will also validate the overall process and show that the mesh used is of a sufficient quality to provide correct results.

The CAD model used represents a simple skin and multiple stringers assembly (see Figure 49) which is used as a stiffener to reinforce the structure.

Figure 50 displays this slat assembly with 7 stringers spanwise (red lines). It is also important to note that the panel is constraint chord wise at the position of the mechanism support. This setup was selected to analyse the deformation of the panel under various loading cases and to see how the panel would react. For further study it will be important to make sure that the skin at the top of the assembly does not separate too much from the main airfoil as this would cause further drag.

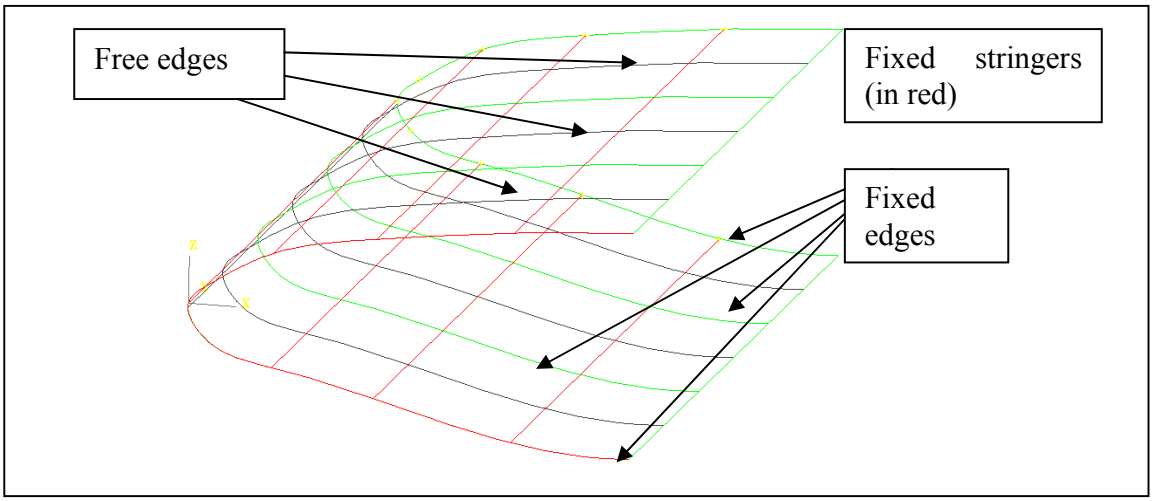


Figure 49 - CAD model Skin & Stringers [Lajux]

The next picture is the representation of the slat assembly under the loading specified above (see Figure 50). It is possible to see where the main stresses are located within the component.

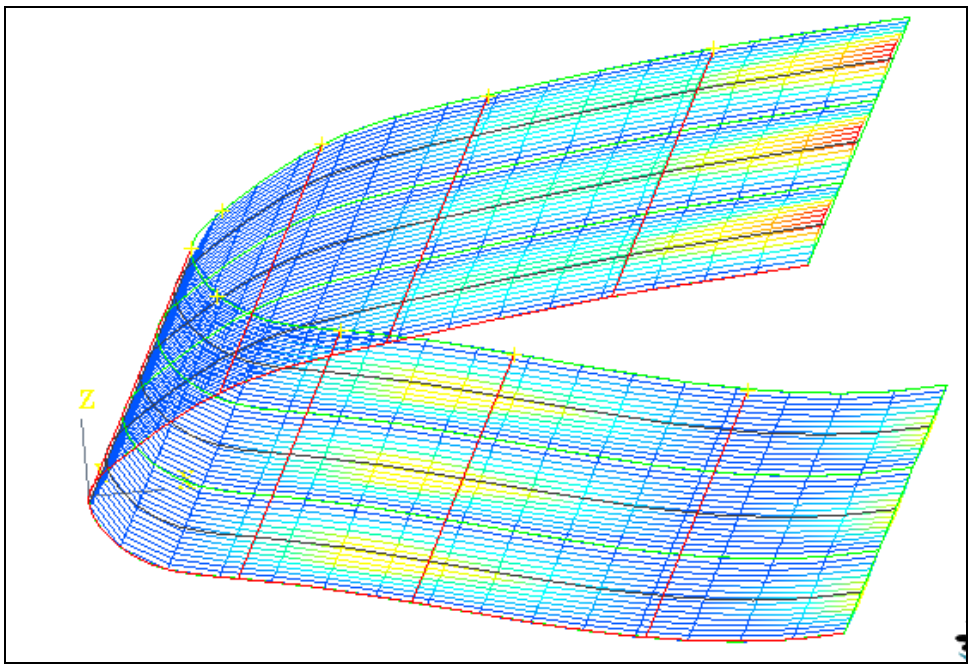


Figure 50 - Slat Assembly nodal displacement under loading case [Lajux]

The following graphs display the difference for the maximum stress (see Figure 51) and the maximum displacement (see Figure 52) for the slat assembly described above. This slat assembly is subjected to a constant load as a pressure applied all over the component ($-0.006 n/mm^2$). The mesh is then being refined to find out at what point this mesh is refined enough and do not need to be refined further.

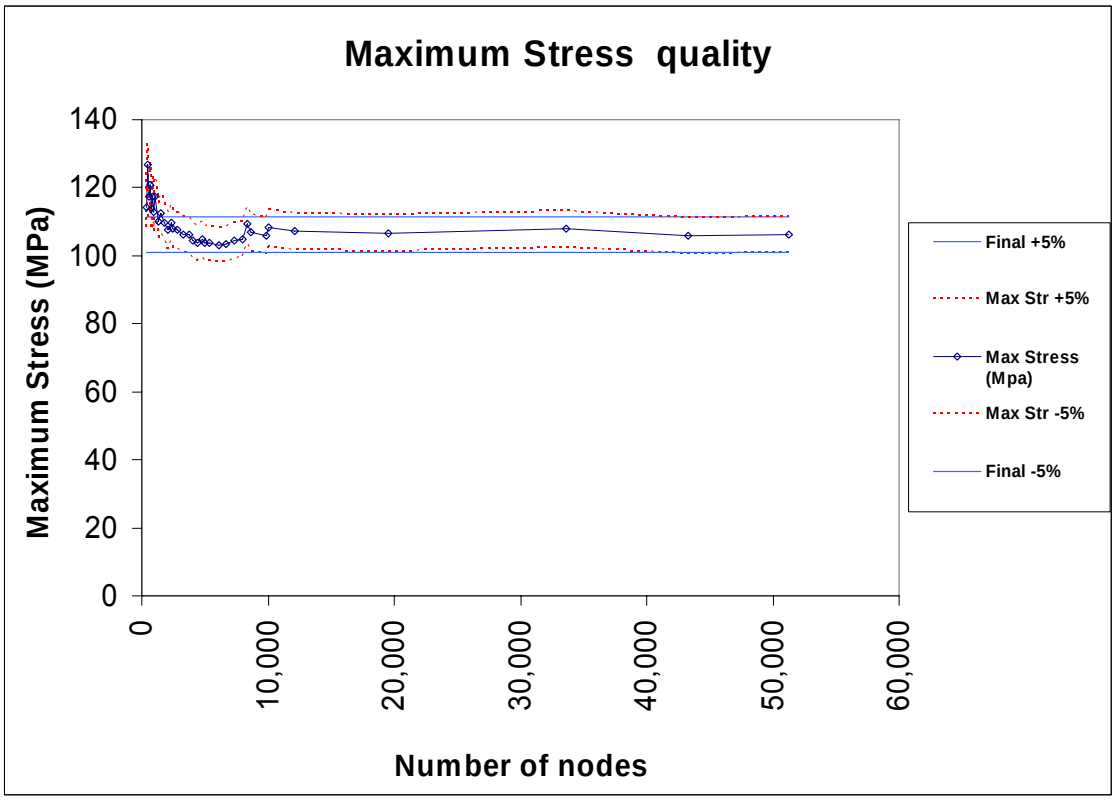


Figure 51 - Maximum Stress [Lajux]

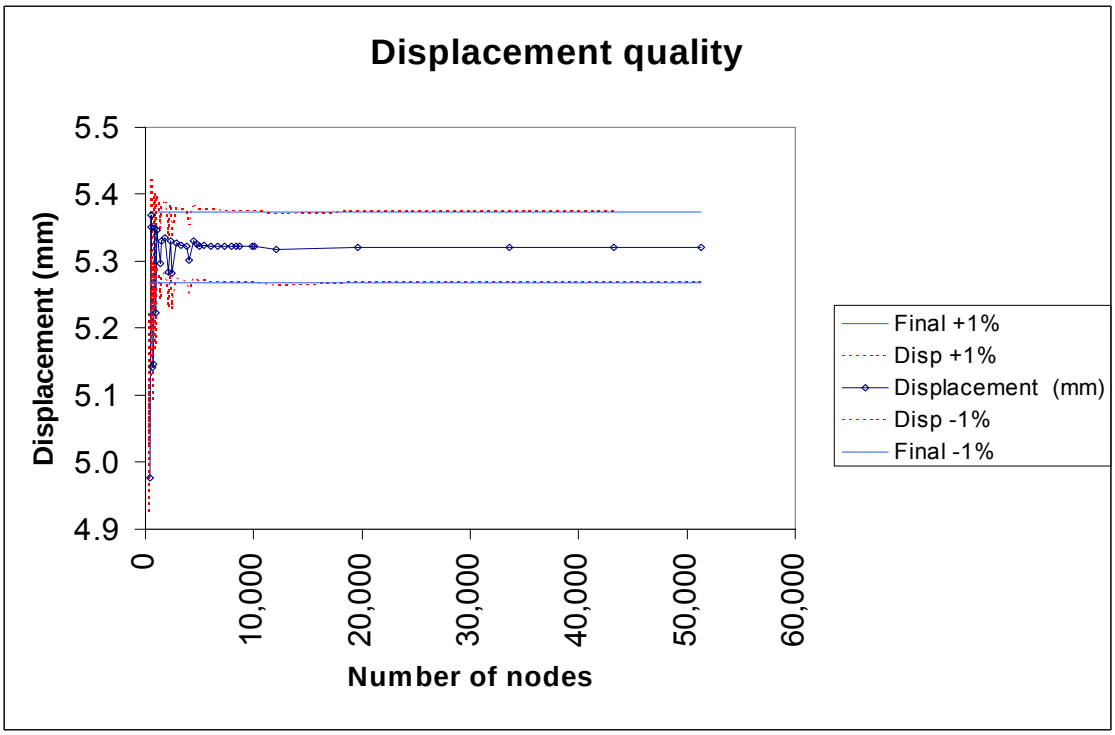


Figure 52 - Maximum Displacement [Lajux]

The two figures on the previous page show that it is reasonable to have a mesh with the number of elements between 10 000 and 15 000 as the values of the maximum stress and the maximum displacement do not vary greatly.

In fact, the value of the maximum stress for a model with 12 500 nodes will be nearly the same, within the 5% range, as a model with 50 000 nodes. However, the model with 35 000 nodes will take something like 2.5h of computing time on a classic PC, when the 12 500 nodes model would take less than 20 minutes!

The same pattern is shown for the result of the maximum displacement compared to the number of nodes in the model. However, using the displacement figure it is possible to see that the difference in displacement is under 1% of the range compared with the displacement obtain with 50 000 nodes model. Due to the very low range of displacement it is accepted that the result obtained with 10 000 nodes is accurate enough.

By combining the two graphs it is clear that a model with 12 500 nodes will be sufficiently refined to provide valid results and still provide the least amount of time spent on computation.

Also, it is important to notice that the meshing method used for this example use a structure mesh with rectangular cells. Even if triangular cells are normally more efficient to mesh a part, for this case rectangular cells are perfect. They are much better for the user to see what kind of quality is the mesh and also to see how well the mesh represents the real geometry.

4.4. Validation of mass and reliability estimation

4.4.1 Validation of mass estimation

In order to validate the author's mass estimation tool it is necessary to compare the result of this tool with the results obtained by other researchers for the same kind of mass estimation. It is also important to use real data from existing aircraft and some of their LE components mass to have a coherent comparison tool.
All this is used together to prove that the mass estimation tool developed for this research provides users with a decent set of results which can be used for further studies. This tool will also be used for the case study (see paragraph 5.4), so once again it is important to ensure the validity of the results.

It was decided to use mass estimation methods developed by the following authors or aircraft manufacturers: Torenbeek[15], Dornier, Grumman and also the Society of Allied Weight Engineers (SAWE)[32].
All of them provide theoretical formulas for the estimation of the mass for different types of LE devices. Most of them developed their formulas to find the mass at the conceptual (or preliminary) design stage to give a fair idea of the LE devices mass. They used data related to the mass of the LE devices for some aircraft, and developed formulas to estimate the mass of these devices compared to the MTOW or the maximum speed of the aircraft. For example, Torenbeek[15] gives a formula to estimate the mass of slat and fixed Krueger flap as a mass to area ratio which depends on the MTOW. On the other side, Grumman[32] only gives a formula for LE flaps (without specifying the type) which depends on the aircraft speed limit.

A comparison of the results from these different estimation methods with the author's method will give a fair idea of the accuracy of this new mass estimation tool.

To validate the results it was necessary to apply the mass estimation tool to different aircraft such as the Boeing B747[41], the Lockheed L1011 [42] or the Hawker Siddeley Trident[43] for which the author had available data regarding the mass of the different components. This provided a set of results as shown in the following graph (Figure 53), which shows clearly the actual mass, and also the estimated mass for each of the different methods. Since the different methods do not always estimate the mass for each type of LE devices, it was necessary to use a mixture of different methods to generate a result which realistically represents the real aircraft system. These estimated results are compared against real data, and also against the author's tool estimated results.

This graph shows the results of the mass estimation for the LE systems which was applied to different aircraft using the different calculation methods. It also compares these results to the actual mass of the LE system of these aircraft. The legend describes which method has been used for the mass estimation (or "actual" for the mass of the LE system from real data), and the bottom axis describes the type of aircraft studied.

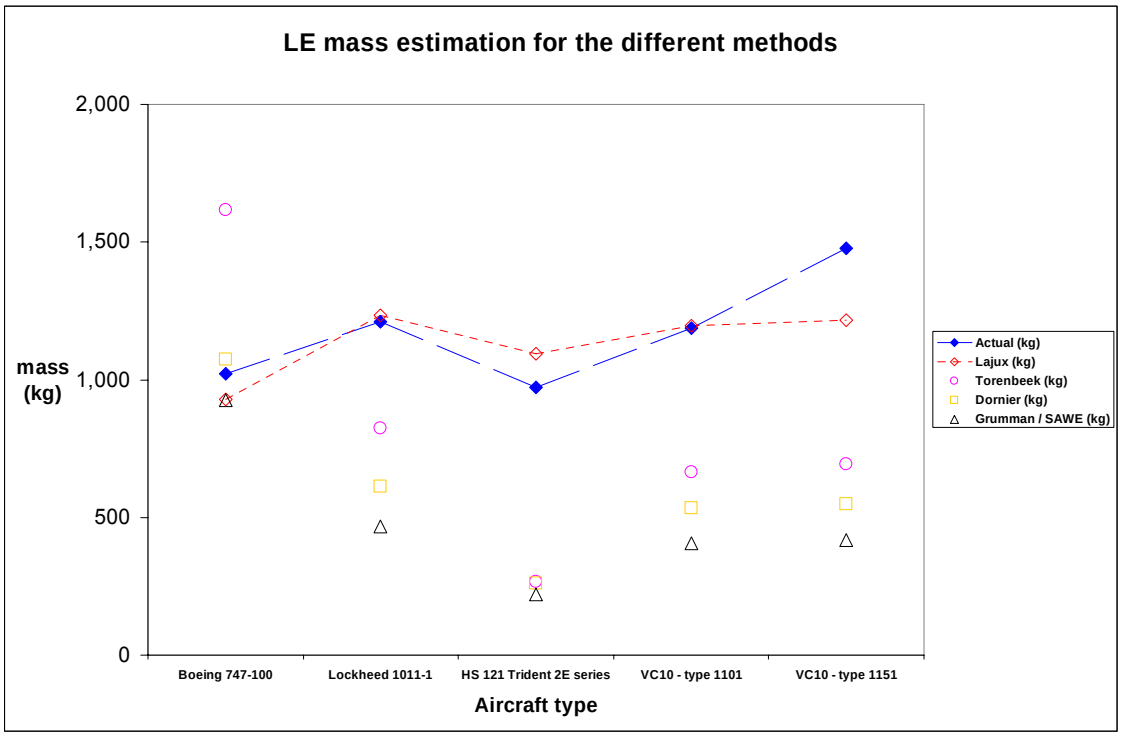


Figure 53 - LE devices mass for different estimation methods [Lajux]

It is clear that the results generated using the newly designed tool for the overall mass are generally closer to the real value than the estimated mass using the other estimation methods. This clearly indicates that the results generated by the authors' tool are generally more reliable than the results provided using the other methods of mass estimation.

This is partially due to the way the estimation is done using the different methods. They generally consider the mass of the LE device being a "portion" of the MTOW or dependant on the maximum speed (as explained before). The results generated by them are generally inaccurate as the overall LE system could have different configurations (geometry, panel area…) and also use different types of mechanisms. This is where the new tool is more efficient as the user can generate a set of results for different configurations. The other mass estimation methods do not give a choice for all the types of devices; they generally describe only one type or the whole LE system (without defining the type of device used).

The newly designed mass estimation tool has more flexibility than any other methods described in the preceding paragraphs. It is also more comprehensive as it has a detailed list of the possible types of LE device. And finally this tool gives more accurate results, so the result will be good enough to use for a case study.

4.4.2 Validation of reliability estimation

The results obtained with the reliability estimation tool have to be validated to insure that this tool generates accurate and coherent results. This validation can be done by using data and results from different references on reliability.

To show the validity of the tool it is necessary to use results for the same type of study and to compare them to the given results. A possible way to do this would be to use data from maintenance data, and maintenance log on different aircraft. Once these data have been collected it would be possible to process them and compare it to the results supplied by the new reliability estimation tool. These sets of results will obviously be about the same aircraft model (or type).

However, due to the lack of information and data for this precise part of the research it was only possible to rely on the proven reliability for non electronic parts [47]. The foundations for the reliability tool parameters are based on the condition of use for the different types of components.

The author also used the research carried out by Bineid [44, 35] on the development of a new method to estimate aircraft maintenance dispatch data and reliability to develop this new tool. In this research, the author gave a new method for the estimation of reliability for different parts of the aircraft. However, this dispatch reliability was done at the aircraft level, and not especially for the LE. So it is clear that it will be necessary to adapt the data from his research to the research carried out by Lajux for this thesis. For this reason the author used some of the reliability calculation methods described by Bineid.

Using the data from his research, and the references used to do the research. It is possible to say that the calculation of reliability for LE devices will be as accurate as possible.

It has to be noticed that the author could not do a complete validation process for this tool. In fact, by reviewing different maintenance logbook it was impossible to see a real trend for the failure of LE devices components. Due to the very low number of noted failure for LE devices it was never possible to make real use of these data.

Also due to the specificity of this reliability tool it is impossible to compare it to other existing tools due to the fundamental difference and specificity of the application.

4.4.3 Validation of the cost assessment

The cost assessment tool is quite similar to the reliability tool from the validation process point of view. In fact, the lack of data (due to industrial confidentiality) for the cost of particular components can to some extend means that the results would be wrong. However, it has to be understood that this tool generate an average pricing structure for different types of LE devices. This way the user can get a detail view of generic cost for different LE devices and can easily compare them. In fact, the validity of the cost estimation tool is in the averaging the cost for different types of devices, this means that the offset cost of making part in machining for example would only be a parameter, and

this parameter would the same of every devices. This means that the difference in the final value could be wrong; however the difference between the costs of two different devices will be correct. This is to do with the cost error (it there is an error) of the initial cost of one parameter. The most important data is the difference of the final cost and not the final cost itself.

The parameters used to generate the cost of LE devices are the ones used by other well known researchers. They normally give an accurate representation of price for different materials per unit of mass, or a "buy to fly" ratio, they are generic value of parameters which cover a broad range of data. They are normally defined using an historical database and averaging empirical data. However, due to the novelty of such a specific tool it is quite difficult to find an accurate method to validate such a tool.

4.5. Summary of the validation results

The objective of this chapter was to demonstrate that the results obtained during the different simulations and computer experiments are valid and could be correlated to existing results, and validated methods.

The other important point was to show that the analysis, both CFD and FEA, have been carried out using a precise validation process. This is including grid independence and convergence of results.

It was also shown that all results, when using computer programs, have been validated in terms of quality, compared to the mesh & solver quality.

For the mass and reliability estimation tools the validation was successfully achieved by showing that the estimated results are more accurate than results generated using other methods. These methods have been developed by well known authors and the validation process used real data for comparing the different methods with the newly developed method.

Therefore it is assumed that the different software (mentioned in this chapter) used for the case study will be accepted as giving accurate results. These results will be valid as it was proven that the overall process was fully controlled during this project.

When it was not possible to obtain an accurate validation, explanations on the reasons why this was not possible were given. Due to the novelty of some of the tools developed for this research, t was sometime not possible to compare it to other research or results.

Due to time and budget constraints it was not possible to test prototypes of the chosen designs solutions. And so it was never possible to obtain more data related to a real model or prototype.

Chapter 5

5. Case Study and Results

5.1. Introduction

This chapter describes the process with which the design methodology is applied to a case study, by taking each part of the methodology and applying it to the chosen test case. The case study for this research was taken from research done by Edi[4] and Ammoo[2] using the Advanced Transport Regional Aircraft (ATRA) as the baseline. This aircraft was considered for this research because of the continuity of work inspired by the research supervisor, and also by the relative availability of data on this aircraft. It is also representative of as an advanced regional aircraft, which is one of the most current designs.The case study presented in this chapter will focus on the design of the LE device (with possible VC), but this chapter will not focus on the use of LE device for high-lift, or HLFC applications. Each part of this chapter relate to the methodology flow chart as the case study will follow the same process (see Figure 34 in page 41)

5.2. Aircraft/case study description

The ATRA (Figure 54) consists of a family of three derivative aircrafts designated ATRA 80, ATRA 100 and ATRA 130 (see Table 7 in appendix A), where the numbers represent the passenger capacity. The ATRA family uses the same airfoil and wing planform, and also features Variable Camber Flap (VCF) and HLFC technology. However for this research the effect of HLFC will be ignored as there will be no suction applied. In this study VCF will be studied primarily for the LE devices.

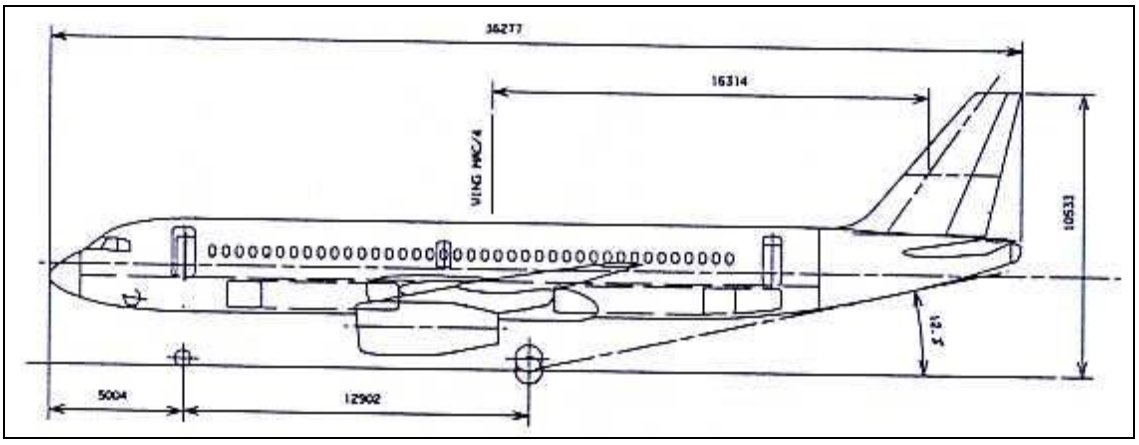


Figure 54 - ATRA 100 [2]

For the case study it has been assumed that the planform and wing area are taken from the ATRA used and studied by Ammoo[2] and Edi[4]. Most of the focus will be on the improvement during the analysis stage due to the availability of analysis tools. Emphasis will also be made on investigating innovative solutions for the deployment of the LE device.

The wing planform is “neutral”, but an attempt will be made to find out what is the best configuration to use on this wing and see how VC concept could be used at the LE of the wing. This will also affect the aerodynamic performance of the ATRA for the different flight phases. All of these things will be studied and analysed in the following pages.

5.3. Inputs

Most of the input data will be taken from the ATRA description. This incorporates the geometry of the wing as well as some of the aerodynamic characteristics Also the author will make assumptions based on experience learned during this research, when necessary.

5.3.1 ATRA wing planform parameters/description

The ATRA wing planform is taken directly from data in Ammoo's thesis[2] which was selected after comparing data from existing aircraft. Only the Aspect Ratio (AR) was taken from the work done by Edi[4].The AR was taken from a sizing chart (as mentioned in chapter 3) with the AR being the crossing point between wing loading and thrust loading and to get the optimum AR for best aerodynamic performances.

The wing planform parameters are as described below:

- ¼ chord sweep = 25 Deg.
- Taper ratio = 0.274
- Aspect Ratio (AR) = 9.5
- Wing area (S) = 110.21 m^2
- Inboard section span (inboard=>kink) = 4 273 mm
- Outboard section span (kink=>tip) = 10 192 mm

5.3.2 Wing torsion box

For this research project it has been decided not to change the torsion box, as the change of LE device should not affect the range of loading. The loads applied to the torsion box should not be greater than the ones experienced in the original ATRA aircraft. It is assumed that the initial design allowable loads will still apply to new solutions.

5.3.3 Variable camber definition

At this stage it is only possible to give directions for the range of displacement/rotation to be achieved for the LE system, and also to decide the chord of the device and its centre of rotation.

Variable camber LE parameters (cruise condition:

- Range of motion = 0 Deg /+5 Deg
- Centre of rotation = X =0.828m
  Y = 0 m

Value of the airfoil profile taken at Y(b/2)=0.2

Variable Camber chord for the LE device is at around:

- Max. Chord (% wing chord)= 20 %

The VC option for the LE device will have to help the aircraft achieving the aero performance required in order to comply with the regulation listed in the methodology.

5.4. Initial mass and reliability assessment

5.4.1 Mass estimation

The estimation method, developed for this research project, uses the dimensions of the LE panels, so it is necessary to know the major dimensions of the wing and especially the ones of the LE panels. This data is available in Table 7 (see page 158 and page 162 for **Appendixes C**), and various other dimensions will be calculated. The estimation tool processes these data and provides the designer with a set of results; it is also possible to do a simple comparison between different configurations.
The case study has the following wing dimensions:

S	=	110.21m^2	Span	=	32.400 m
AR	=	9.5	Semi-span (s)	=	16.179 m

Cr =	4.830	m	VC LE device % of wing chord	=	20%
Ct =	1.586	m	Standard Mean Chord	=	3.400m

To estimate the mass of the LE systems for the ATRA aircraft it is necessary to establish what will be the area of the inboard and outboard sections. To find this area, it is necessary to find the exact span and chord of these two sections:

	Inboard section:	Outboard section:
Span=	3.308 m	8.263 m
Chord=	1.158 m	0.317 m

Possible number of panels:

Inboard= 2 / 3	Outboard= 4 / 5 / 6

2=> Panel Span =	1.654	m	4=> Panel Span =	2.065	m
3=> Panel Span =	1.102	m	5=> Panel Span =	1.652	m
			6=> Panel Span =	1.377	m

The preceding dimensions were obtained by taking 80% of the wing span (or available span) for the LE device, which is the general amount of span dedicated to the LE device in modern aircraft. The remaining 20% of the span is generally made up of a fixed LE and the fuselage region.
For the chord it has been decided to use 20% of the airfoil chord, as it represents the position of the front spar, and ultimately the amount of chord dedicated to the LE device would not go all the way back to the LE front spar.

Initially it is necessary to find the mass estimation for different types of device in the case study in order to find the most suitable one, which is likely to be the lightest option, as well as providing the expected aerodynamic performance.
For this, it is necessary to generate a set of results depending on the different possible configurations for the number of panels and type of devices (Figure 55).

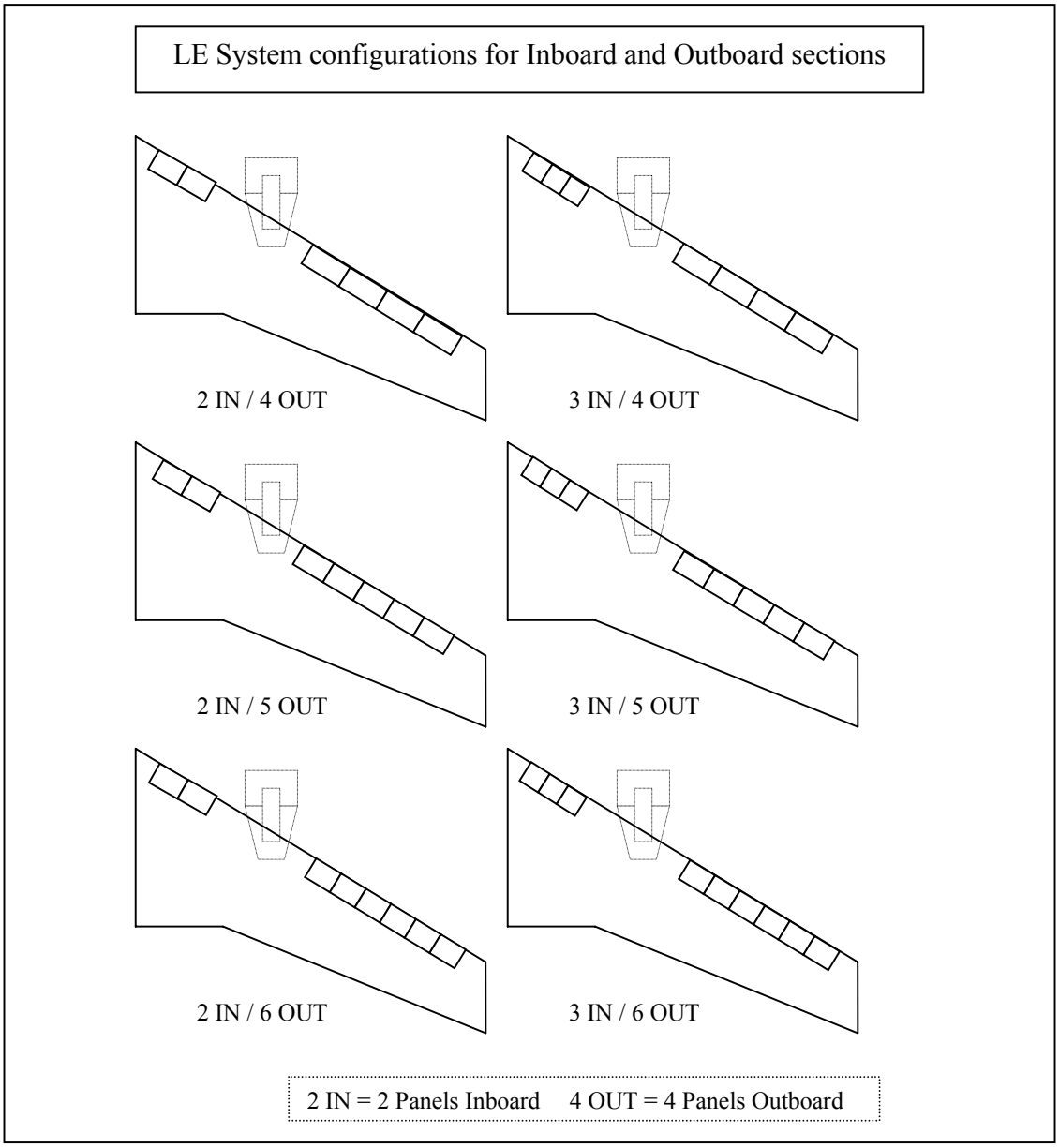


Figure 55 - Different wing layout configurations [Lajux]

The following graph (Figure 56) shows the mass of the LE system for one wing only. The different configurations possible for the inboard and outboard sections of the wing are shown in Figure 55. Figure 56 displays the mass result with the different outboard settings on the bottom and the different inboard settings being described in the legend. This displays the set of results for all the different combinations possible, using the mass estimation tool. Obviously all these data have been generated using the parameters for the wing and the section geometry as described before. It was decided to estimate the mass using only one type of device per section as It would be too costly and complex to have two types of devices within the same section. This is possible because this tool covers the full range of LE types, but to some extend not realistic or practical.
Using these parameters with the new program gave the following results:

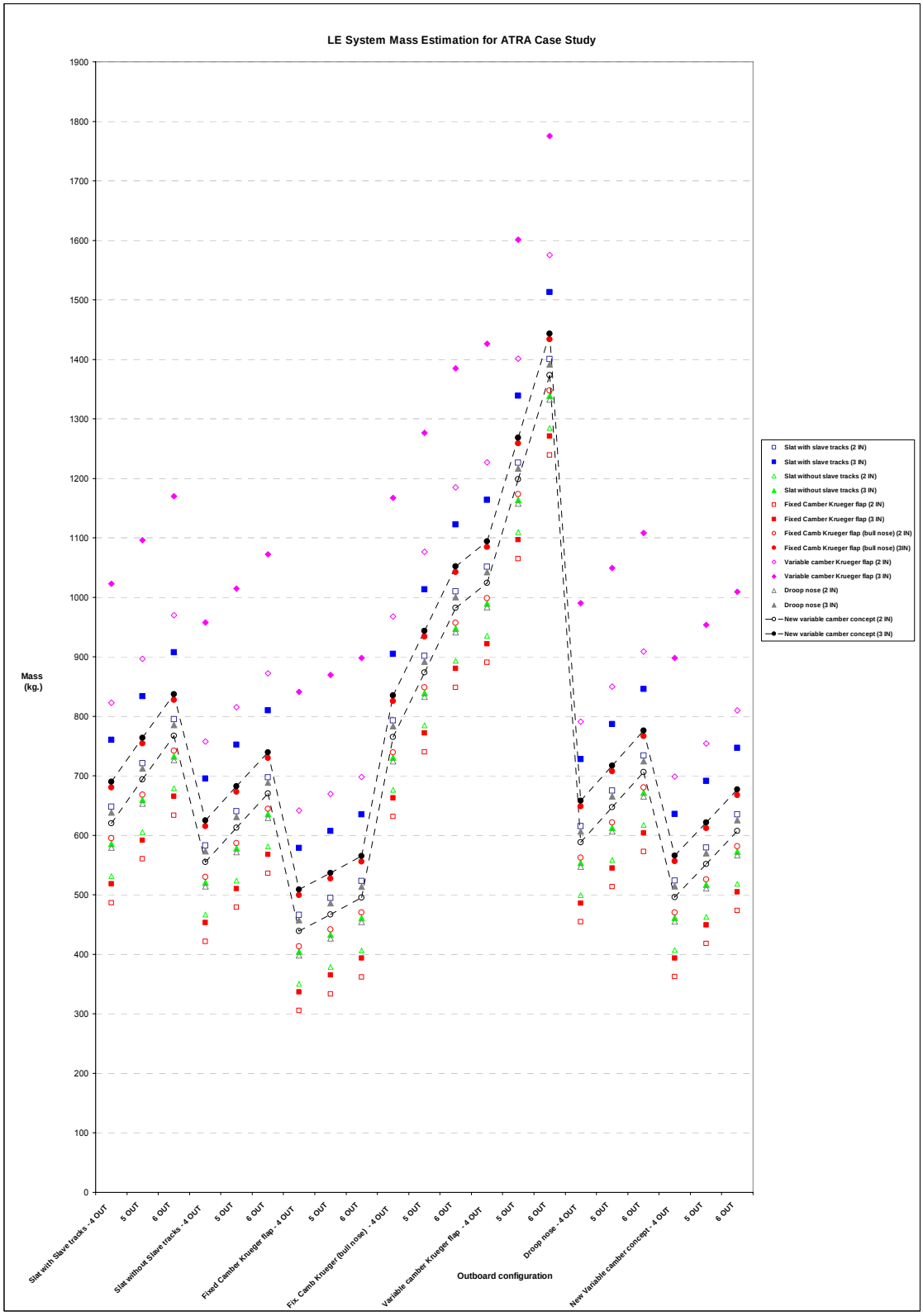


Figure 56 - LE device mass (1 wing - different configurations) [Lajux]

This graph (Figure 56) shows that the effect of having a VC LE device is more or less in the middle, compared to the other device masses for the same panel configurations. Having a VC device for the LE does not appear to have a massive effect on the weight of the overall aircraft. It is also important to notice that the Krueger VC device appears to provide variable camber but for low speed operation only it appears to be considerably heavier than a VC LE solution.

However, for a mass which would be generally around 30% lighter than a VC Krueger flap, which is not used at cruise condition, a new VC concept could be used as it is lighter and provides better performance than the VC Krueger concept.
On the other hand, a new VC concept might prove 30% heavier than the fixed camber Krueger, which is light and also easy to manufacture. But once again for this concept there is not much advantage in terms of aerodynamic performance as it is not used during cruise. However the slat and VC configuration seems to provide more or less similar results.

Looking at the preceding graph it is clear that the lightest option would be to have a fixed Krueger flap along the wing with the minimum number of 6 panels (2IN/4OUT) for a total mass of 305kg per wing. However, as explained above, such a device is only used on take off and landing to provide more lift. Compared to the lightest option the most critical case for the mass of LE is achieved using the VC Krueger flap, as the mass reaches a maximum value of 1084kg per wing (for 3IN/6OUT panels' arrangement).

So it is decided that the case study can use the new VC LE device (but not a VC Krueger).

The arrangements, including a set of VC LE devices inboard and outboard, have an estimated mass ranging from 496kg (for 2IN/4OUT panels) to 677kg per wing (for 3IN/6OUT panels). All the other panel arrangements for the VC LE device option have a mass between these last two values. Figure 57 shows clearly the mass for the different configurations listed above; these are the most critical configurations (lightest and heaviest) as well as the chosen configuration for the case study. Also, it was decided to show the result for the more classical slat track concept configuration. It appears that the VC LE device would in fact perform quite well as it is expected to be slightly lighter. It is planned to use the VC effect during cruise as well as during landing and take off in order to improve aircraft aerodynamic performance, but that will be analysed at a later stage in this report.

For this reason, it was decided to select a panel arrangement of 2 panels for the inboard section and 4 panels for the outboard section, as it was the lightest option. Both sections will use VC technology. This choice is made because it has been shown that using VC technology should have a positive impact on the LE system mass compared to other types of devices, as well as being suitable for cruise conditions.
However, this choice will also depend on the reliability of such devices. In the case that it proves to be unreliable compared to the other devices then the final choice may have to be different.

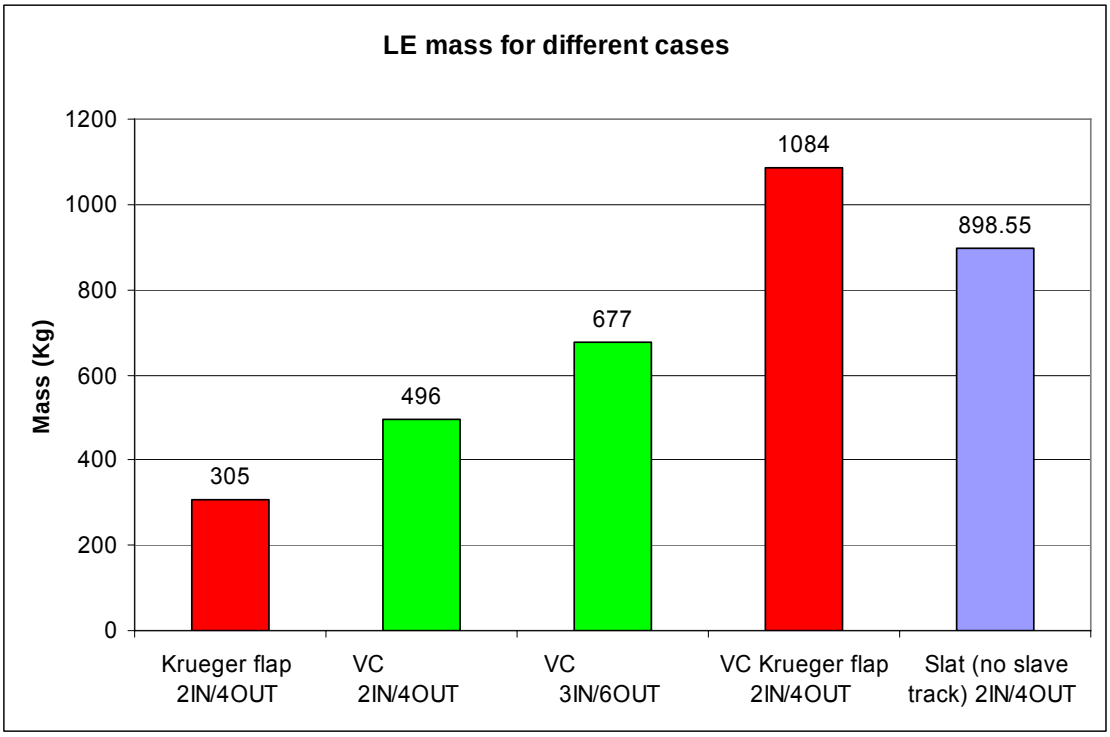


Figure 57 - LE masses for Case Study [Lajux]

To cover such a possibility, some alternative options were selected in order to find the best arrangements when matching the mass and the reliability estimation results. These alternative possibilities can be found on the graph by drawing horizontal lines, with the top one being at 677kg and the bottom one being at 496kg (as cited above).
The following configurations lay within the range of mass [496kg; 677kg].

Config.	*INBOARD (nb of panel)*	*OUTBOARD (nb of panel)*
1	2/3 Fixed Camber Krueger	4/5/6 Slat with slave tracks
2	3 Slat with no slave tracks	4/5/6 Slat with no slave tracks
3	2 Droop nose	4/5/6 Slat with no slave tracks
4	2 Fixed Camber Krueger (Bull nose)	4/5/6 Slat with no slave tracks
5	3 Slat with no slave tracks	4/5/6 Fixed Camber Krueger
6	3 Fixed Camber Krueger (Bull nose)	4/5/6 Fixed Camber Krueger
7	3 Variable Camber LE	4/5/6 Fixed Camber Krueger
8	2/3 Slat with no slave tracks	4/5/6 Droop nose
9	2 Droop nose	4/5/6 Droop nose

Table 5 - Alternatives configurations for mass estimation [Lajux]

Looking at the above table (Table 5) it is possible to comment on the possible configurations as alternatives for the VC LE concept option. In fact, it is not common to use a Krueger (fixed, VC or Bull nose) flap on the outboard section as this type of flap normally requires a thicker airfoil. Krueger flaps are often used to inboard of the wing to guarantee root stall. So, configurations 5, 6 and 7 should be avoided.

Moreover, the droop nose solution, like the Krueger flap, might require a thicker airfoil as it is generally larger and located in the inboard instead of the outboard (see the Airbus A380). However, the droop nose option could be fitted into other airfoil as it is used on thin airfoil for fighter jets. So configurations 8 and 9 might be possible but should be avoided as they appears to be a bad design choice for this application due to the airfoil thickness and also because they are normally at the inboard to guarantee inboard stall.

Configurations 1 to 4 are the most likely to be used as alternatives to the VC LE solution as they show to have acceptable mass and also because they are good design choices for this case study. These configurations will be used in the next stage for the reliability study to see if any of them are better overall than the VC LE concept, using the lightest option of 2 panels inboard and 4 panels outboard.

Regarding the newly designed mass estimation tool, it has to be mentioned that it does not include a change of mass ratio when changing the number of mechanisms (or tracks). This means that the result for a higher number of tracks might not be exact. It is also important to note that this problem is the same for each kind of device and so it is clear that even with more tracks the rate of change between the different devices will not change[29]. Future work should examine the mass changes to slat structures caused by reduced or increased numbers of supports.

The next paragraph will show the estimation of the reliability for the ATRA aircraft, as the results of the mass estimation must be compared with the reliability results to select the final configuration, assuming that they all meet the aerodynamic requirements Figure 70 in page 117.

5.4.2 Reliability estimation

The reliability estimation is calculated using the author's new tool to find the reliability of the overall system for different configurations of the wing layout for LE devices. It is also possible to use this program to quickly generate an approximation of the aircraft life (flying hours) depending on the conditions of use (long, medium or short haul). This enables the estimation of relative reliability per flight cycle.

By trying different configurations it is possible to obtain different sets of results and see which one of these configurations will be the most reliable.

Figure 58, in page 100, shows the result of the estimated reliability for the different settings on one wing. The calculation process used can be found in **Appendixes C** It gives the failure rate per 1 million hours of use (Y axis), as a measure of reliability for the different possible configurations. Lower rates mean better reliability for the given configuration.

It was decided to display the results using one kind of device for each section (inboard and outboard) to keep to the same format as for the mass estimation. This was done for the different numbers of panels for both sections in order to show the overall set of results. This graph can then be used to find the most reliable setting for the ATRA aircraft.

The bottom line represents the different configurations for the outboard section of the wing. The different inboard configurations are represented on the legend using different symbols and colours.

For this analysis, by looking at the graph (see Figure 58) it shows clearly the disadvantage of increasing the number of panels. This increases the number of separate parts and so increases the failure rate of the overall system.
Overall the configuration using the VC LE option looks to be more reliable than the others (see Figure 58). In fact, the low failure rate of the configurations using the VC LE option lay between 1 330 failure/10^6 hours (for the 2IN/4OUT panels' configuration) and 1 996 failure/10^6 hours (for the 3IN/6OUT panels' configuration). The failure rate of the other VC LE panel configurations also lies between these two values.
However regarding the reliability estimation, it was decided to select the configuration using 2 panels inboard and 4 panels outboard, all of them using a VC LE concept.

As with the mass estimation, it was necessary to establish the alternatives solutions for the reliability estimation. These configurations were compared with the mass estimation results so that a final decision on the optimum configuration could be made.
The alternative configurations are defined in Table 6. They were selected by using the maximum and minimum values of the reliability for the VC LE options. By tracing an imaginary horizontal line between these two values it was possible to find the configurations which had an overall reliability falling within the same values as the VC LE options.

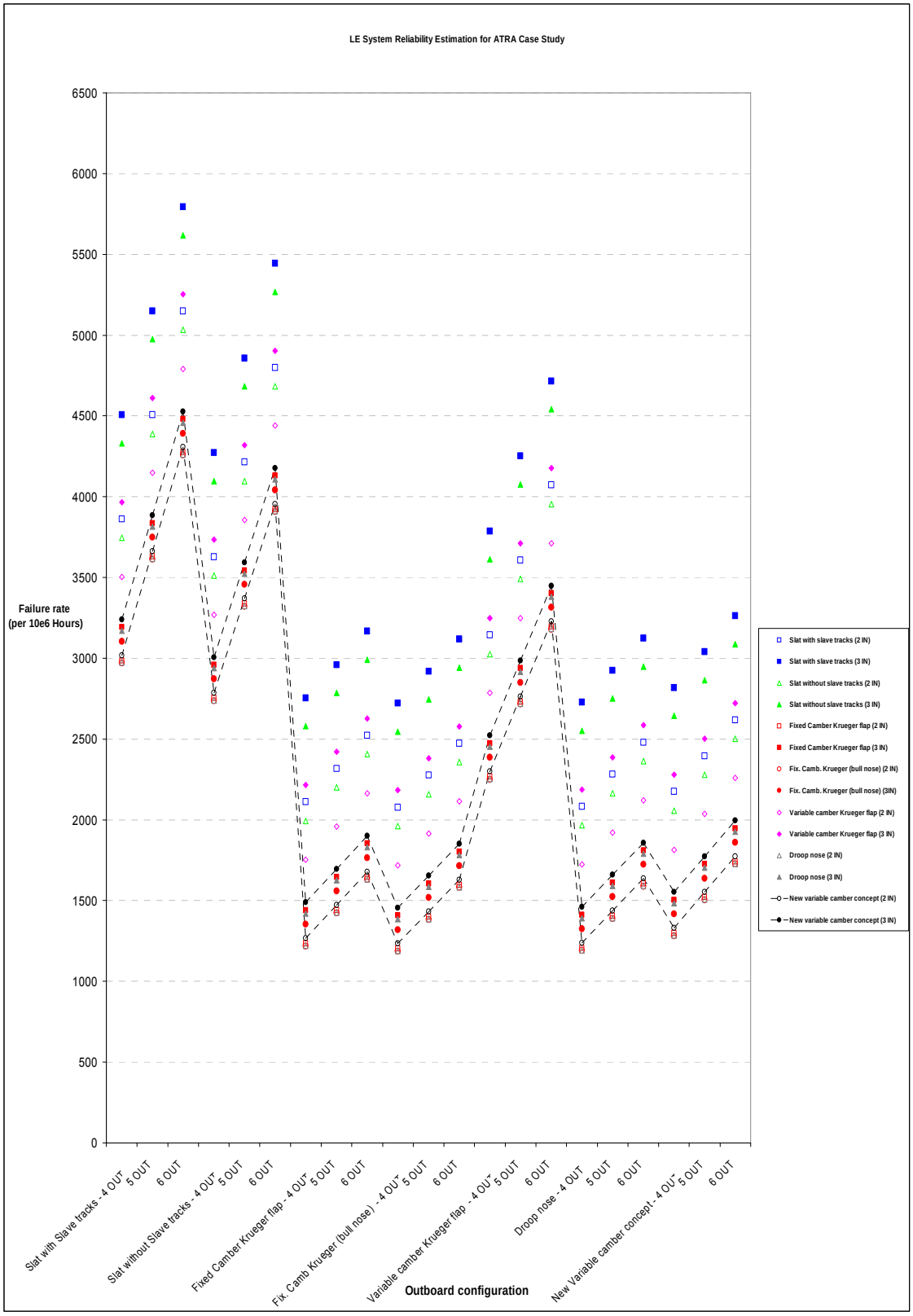


Figure 58 - LE failure rate estimation (different configurations) [Lajux]

Config.	*INBOARD (nb of panel)*	*OUTBOARD (nb of panel)*
1	2/3 VC LE device	4/5/6 Fixed Camber Krueger
2	2/3 Fix. Camb. Krueger (bull no.)	4/5/6 Fixed Camber Krueger
3	2/3 Droop nose	4/5/6 Fixed Camber Krueger
4	2/3 VC LE device	4/5/6 Fixed Camber Krueger (Bull nose)
5	2/3 Fix. Camb. Krueger (bull no.)	4/5/6 Fixed Camber Krueger (Bull nose)
6	2/3 Droop nose	4/5/6 Fixed Camber Krueger (Bull nose)

Table 6 - Other alternatives for reliability [Lajux]

This table was compiled to show the equivalent reliability for other configurations; however this did not take into account any engineering decisions, or the aerodynamic efficiency of each type of device. The proposed solutions, as alternatives, do not always give appropriate solutions.

Indeed, the configurations 1 to 6 all have a fixed camber Krueger flap (with or without bull nose) at the outboard section. This is not ideal, especially as the bull-nose concept requires an even thicker wing profile, which is normally located in the inboard section. And also this type of device is not deployed during cruise as it only has two positions: stowed or deployed. So configurations 4 to 6 should be avoided because of the loss of performance and the complexity of engineering such a mechanism.

The only configurations which can be selected should be configurations 1 to 3, as they all use appropriate devices for the inboard section and use a simple fixed camber Krueger flap on the outboard.

Using the same estimation tool, it is possible to see what effect the reliability would have in terms of cost depending on the different solutions. It is also possible to simulate the effect of different aircraft life time, and see the financial impact on passengers fare (this depending on the number of seats). This analysis has been done using an average cost for one repair.

5.4.3 Cost estimation

The cost estimation for this case study covers only the different configurations short-listed during the mass and reliability estimation. From there the cost is estimated to see the overall impact and what might be the best configurations to use cost wise.

Following on the results of the mass and reliability estimation it is expected that the cost of maintaining the devices or carrying the extra mass might show a similar trend due to the relation between the earlier estimation and the cost estimation. However, the cost estimation tool also considers the cost of manufacturing the part, including the different "buy to fly" ratio depending on the material used or the manufacturing process. Also the initial cost of purchasing the material in the first place might actually show a different trend in the expected results.

The configurations considered for this cost analysis are the ones selected on the mass estimation:
VC LE devices inboard and outboard have 2IN/4OUT panels or 3IN/6OUT panels

Also for the reliability estimation the preferred configuration is:
VC LE devices inboard and outboard have 2IN/4OUT

All these options are considered alongside the specified configuration used for the case study.

The graphs in Figure 59 and Figure 60 show the estimation cost of manufacturing the different sets of LE device configurations both for the inboard side and the outboard side. These two graphs shows the differences between the specified case study configuration on the left (in green) and the cost of the optional configuration (In blue)

The optional configurations come from the mass and reliability study as optional choices if the VC LE device were not used.

As far as the cost of manufacturing is concerned, the use of the VC LE option does not show to be the worst option. In fact, the VC LE device option appears to be the medium cost option for both the inboard and outboard sets. That shows that it would be possible to use the VC LE as a possible device as this configuration and it will not penalise the airlines as part of the aircraft-buying process. In other words, for this case study the VC LE option is not critical, it will increase the purchasing price but should offer higher aerodynamic performance than the cheaper to manufacture configurations. See the aerodynamic paragraph for more details on this (see paragraph 5.5 in page 106).

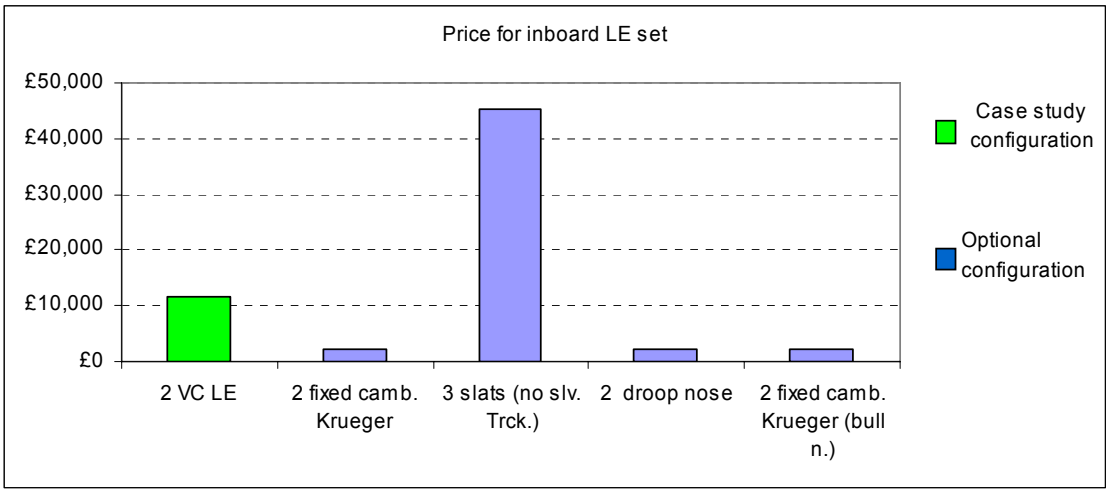


Figure 59 – Manufacturing cost for LE Inboard sets [Lajux]

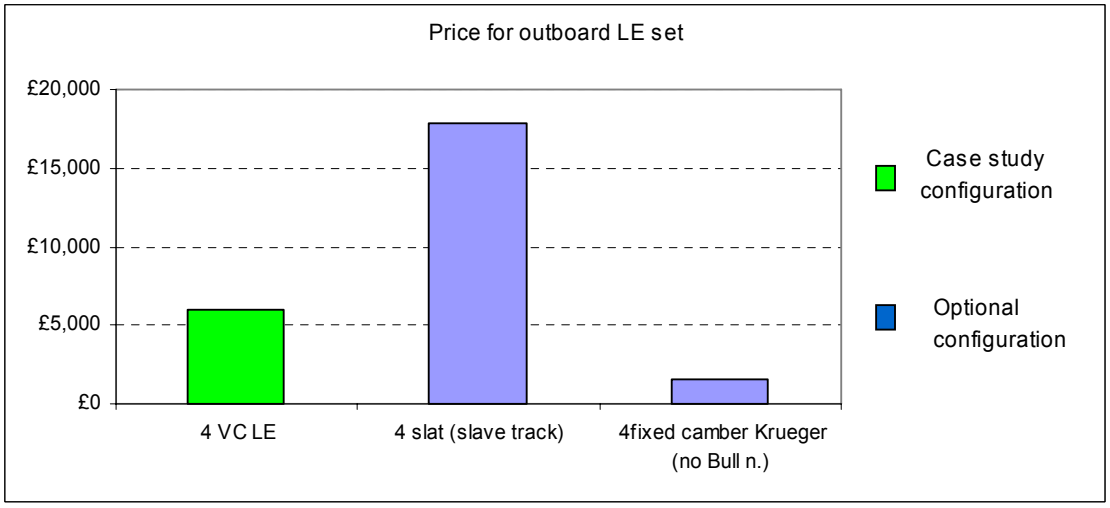


Figure 60 – Manufacturing cost of LE outboard set [Lajux]

Following on the cost of manufacturing, it is necessary to see the maintenance cost implications on an overall fleet. As explained in the methodology, it is likely that an airline will use a fleet and so the airline will need to have an idea of the cost as well as the frequency of maintenance for the LE device. By using one of the tools developed by the author it is clearly possible to estimate the average maintenance cost depending on the type of LE used as the number of aircraft in the fleet.

As explained in the first paragraph of this chapter the ATRA is a regional aircraft and for this reason the author decided to apply a short haul flight pattern to a simulated fleet of 50 aircraft.

Using the different configurations selected, the one for the case study and the ones for the optional configurations, it was possible to generate a set of results to estimate the financial cost of the maintenance per month for the overall fleet.

The results of this analysis are presented in the graph below (see Figure 61), it shows the breakdown for the cost of the different configurations. The process for the calculation used for this case study is explained in the appendices in page162

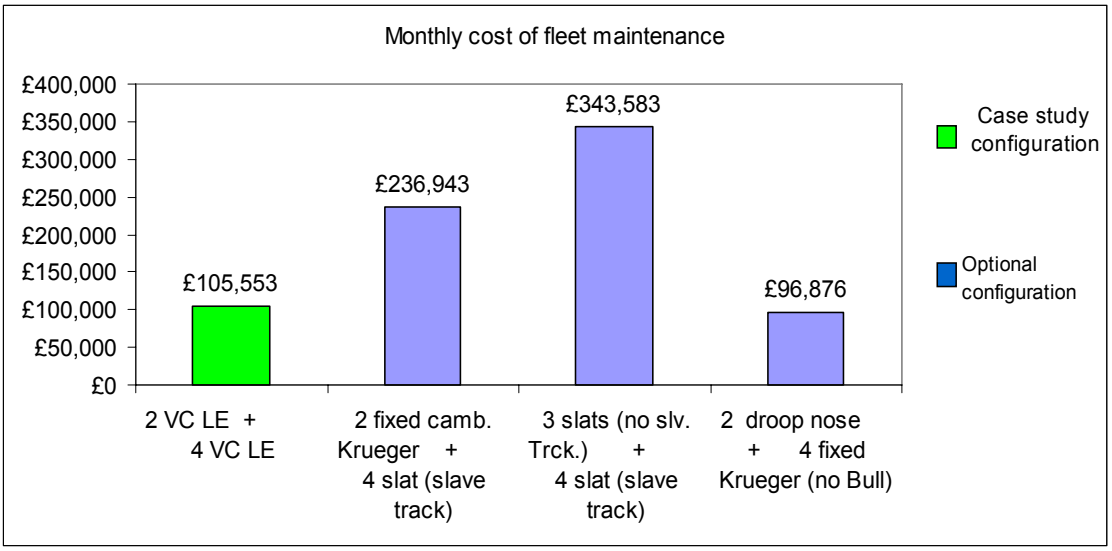


Figure 61 - Monthly cost of fleet maintenance [Lajux]

From this graph it is clear that the cost of maintenance for the VC LE devices to be used on the ATRA will be quite low compared to the other configurations. In fact, the case study configuration has more or less the same monthly cost as the cheapest option.
Once again it is shown that the VC LE devices could be a potentialyl good configuration as it is one of the cheapest to “run” (or to maintain).

5.4.4 Results for mass, reliability and cost estimation

In order to find the optimal and most efficient configuration for the ATRA case study it was important to consider all the possible alternatives as well as the optimum solutions specified by both the reliability estimation, the mass and cost estimation analysis. These analyses have given solutions as well as providing comments related to the engineering knowledge and the usefulness of the different configurations. Some of them were giving a lower mass or a higher reliability but were not necessarily useful for this case study.

It appears that both analyses have shown that using the VC LE option with the minimum number of panels is an optimum solution, using the configuration with 2 panels inboard and 4 panels outboard.

The alternative solutions provided by the mass estimations appear to be a lot less reliable than the VC LE option. Using Table 5 (see page 97) it is possible to see the alternative configurations for the mass estimation. Configuration 1 seems to be between 2 and 5 times less reliable, depending on the number of panels, than the optimum VC LE configuration. The same applies to configuration 2 which is more than 5 times less reliable than the chosen optimum configuration.

The alternative solutions provided by the reliability estimation appear to be heavier than the VC LE option. Using Table 6 (see page 101) it is possible to see the given alternatives for the reliability estimation. Regarding the mass estimation results configurations 1, 2 and 3 are around 100 kg heavier than the optimum configuration. Although they are heavier, they can still be considered as a possible alternative, due to the low mass increase.

However, the simplest configuration, using the VC, has been selected to be applied for the case study as it has been proven that it is the optimum solution. The final configuration has 2 panels for the inboard section and 4 panels for the outboard section. Both sections will be using a VC LE device which will be designed and analysed in a later part of this report.
The different estimations proved that using a VC LE device concept could be a suitable design solution to use on the ATRA aircraft, as both estimations showed good results for this type of device. Also the VC LE device is expected to provide better in flight performance than the more classical concepts, without being too heavy or too unreliable. Also the cost of making and maintaining such a device does not appear to be a burden compared to other proposed configurations. Overall this concept seems to be a good solution.

The dimensions of the LE panels (all using a VC LE device) are as follows:

	Inboard section:		Outboard section:	
Span=	3.308	m	8.263	m
Chord=	1.158	m	0.317	m
Nb of Panels =	2		4	
Panel Span =	1.654	m	2.065	m

5.5. Aerodynamic design

For this research the analysis of the aerodynamic performance is restricted to a theoretical study. This was done using a CFD package (FLUENT) and XFOIL. Due to the scope of this project and the budget constraints it was not possible to use prototypes for wind tunnel and structural testing.

The ATRA aircraft was designed to operate at a maximum speed of Mach 0.82. The objectives of the ATRA wing are to have a nearly two dimensional flow (and with a near no sweepback effect on Mach number), but also to keep the greatest amount of laminar flow possible (improved L/D), and finally to achieve good performance in off-design operations [4].

It is also important to know that the outboard wing is designed to have a constant aerofoil thickness ratio (t/c), in order to have a constant shock position and strength as well as a constant section lift. This wing was also designed to sustain laminar flow to 55%, or more, of the upper surface, but this was including the effect of the suction device of the HLFW. As this study does not consider the use of a HLFW the overall performance of the wing will be altered. This alteration of performance might be overcome by using a VC device.

Other parameters set for this wing are little or no flow separation at Mach 0.8 (cruise speed), a wing lift coefficient of 0.5 and 25deg quarter chord sweepback.
This wing was been used by Edi and Ammoo in their PhD Thesis. They referred to a wing outboard section from a Boeing technical report [45]. This aerofoil was said to be typical of that used for the wing outboard section and also closely matched with the specification set for the ATRA 100 wing.

The inboard section is in fact a deviation (modified version) of the outboard section. The main changes are a higher thickness ration (t/c) and a longer chord.

For this research the effect and modification of the wing tip are not analysed or studied.

As it has been mentioned in the opening part of paragraph 5.5 the aerodynamic performance analysis prediction for this case study was performed using two methods, Xfoil for the low speed case and a CFD package (FLUENT) for the high speed case.

FLUENT is a state of the art computational program for modelling fluid flow and heat transfer in complex geometries[46]. FLUENT is used to simulate the airflow around the airfoil for this research project. It also supplies data on C_D, C_L and C_p which can be used to analyse the aerofoil performance.

Speed calculation for airflow (calculation of Mach number):

$$c = \sqrt{\kappa \times R \times T}$$

Equation 5-1

c=	Speed of sound	[m.s^{-1}]
κ=	Adiabatic index (1.402 for air), it is also called γ	
R=	Universal gas constant (287.05 J/kg.K)	[J/kg.K]
T=	Temperature	[K]

Alternatively this equation can be replaced by a simpler one using temperature in degree Celsius instead, but this formula is not as precise at the precedent one:

$$c = (331.5 + 0.6\ \vartheta)$$

Equation 5-2

c=	Speed of sound	[m.s^{-1}]
ϑ=	Temperature (theta)	[C°]

All the aerodynamic results are explained in the appendices D.

5.5.1 Aerodynamic requirements

The aircraft used for this case study has to comply with the regulations for the regional subsonic aircraft, with typically only one engine on each side of the wing. The requirements for the different phase of the flight are found in the documentation provided by the aviation authorities.
It is expected that the generated solution for this case study will do better than the original airfoil. This will be verified during the aerodynamic analysis.

5.5.2 Geometrical constraints

The airfoil generic shape has been fully described and so it is apparent that there are constraints in term of space to fit the LE deployment mechanism.
The front spar will be located at 20% of the chord, from the leading edge of the airfoil; this will allow enough space to package the deployment mechanism. The type of device to be fitted has been decided at the mass and reliability estimation stage.
The maximum thickness of the wing has been defined using the airfoil selected for the cruise condition. This gives the slimmest airfoil possible and so the worst case scenario for the available space for the LE device.
It is also important to know the location of any fuel tank, as it could be necessary to use some of the fuel tank space to fit a track can, if tracks have to be used.
At this stage of the study it is also very important to know if the LE device will have any chordwise extension as this will have a direct impact on the way the LE device is deployed.

Geometrical constraints:
Front spar = 20% of chord
Maximum thickness= 0.441m (at cruise condition)
Chordwise extension= NO.

5.5.3 Initial flap airfoil

The initial flap airfoil (or generic airfoil including the leading edge) can then be generated using CAD tools. This airfoil is created using the data from the previous paragraphs. Using this profile the user start setting the front spar at 20% of the chord as specified earlier in this document. By having the front spar in place, and also the airfoil profile it is possible to start the generation of the different camber position (from 0 Deg. to 5 Deg). Figure 62 in the following page shows an example of LE section it is possible to obtain at this stage. Also the user can simplify the representation of the spar by making a simple vertical line. This type of section view can help designers to consider the different issues linked with the given profile and section. These issues will be likely to be the envelope for fitting the deployment mechanism as well as the possible position for the fixing points.

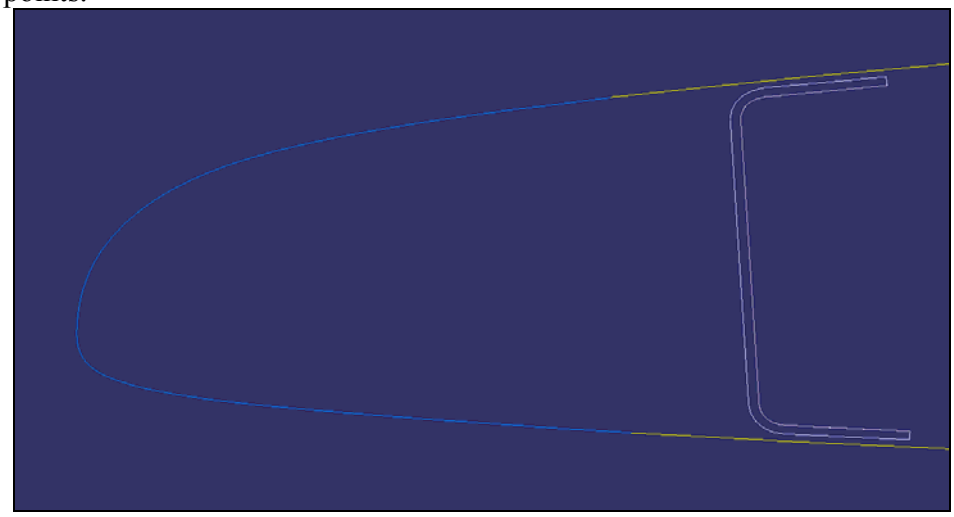


Figure 62 - LE section view [Lajux]

This airfoil has different settings using the possible VC at the LE; it is possible to generate the airfoil for each deflection needed for this study using a tool developed by the author and also to generate the different profiles. A graphical representation of the different section profiles will be shown in one of the following paragraphs, in page 119. These profiles will then be used for the mechanism design.
These settings and positions will be used at a later stage for the design of the mechanism.

5.5.4 Layout configuration

The overall LE system layout is defined using the data from the mass & reliability estimation which gave the optimum configuration for the design of the LE panels. However, the overall layout has still to be defined in more detail to see clearly where the "moving" panels are located on the wing (spanwise).

First of all referring to chapter 5.4 , it has been decided to use 2 panels for the inboard section and 4 panels for the outboard section (see Figure 63 for illustration).
Another point to take into account is the engine position and the nacelle, as both these items have a direct impact on the possible space to deploy the nearby LE panels. On the ATRA there is only one engine on each wing and so the LE panels will be split around the engine, the inboard section being located between the fuselage and the engine, and the outboard being located between the engine and the wing tip. This is also important in the case of possible chordwise extension of the LE during deployment.
It is important to think about possible design solution to keep the spanwise continuity for aerodynamic purpose. So during the mechanical study phase, this point will have to be taken into account.

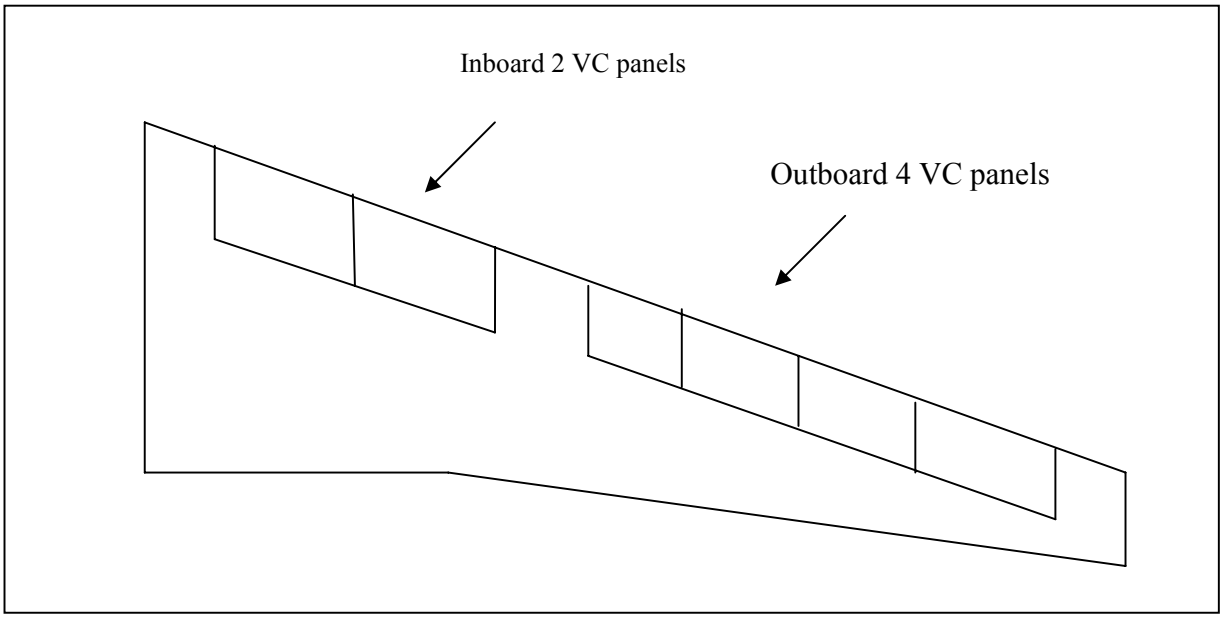


Figure 63 - LE lay-out configuration [Lajux]

Overall it is important see that no components are clashing together during the different flight phase as the LE panels are deployed and stowed. It is compulsory to have a totally clash free space to allow the full range of motion of the different LE panels and components.

5.5.5 Check aerodynamic performance

In order to obtain the detailed results regarding the aerodynamic performance of the wing, also including the VC at the LE, the Fluent CFD software and the Xfoil programs have been used. The Xfoil software was used to test the wing profile at low speed due to the inviscid flow condition in the software. Fluent was used for the study of the airfoil behaviour at cruise condition, as this program takes into account the effect of viscosity.
Each process and the result of the analysis are described below.

As explained above the low speed simulation was done using the Xfoil software, this software allowed to generate sets of results for the Cl, Cd and L/D ratio to define the optimum aerodynamic characteristic. Different cases were run to analyse the case study airfoil profile under different conditions and also using different camber angles. As specified in precedent paragraph it was decided to use a LE device with a specific chord being 20% the airfoil chord. This way it is possible to show the performance of the VC concept for the LE device.

⇨ **Low speed case**

The results of the Xfoil analysis at low speed are shown in the following figures (see Figure 64 and Figure 65). These figures show that L/D ratio for the selected airfoil, this easily shows the airfoil will perform against angles of incidence, as well as under different speeds.

Using these graphs, it is clear that the LE device increase L/D for high angles of incidence. This is especially true for the range of angles of incidence usually used for take off and landing. That implies that the VC LE device actually improves both the landing and take off performance by increasing the lift and reducing the drag at the same time. There is a clear offset of the L/D created by the LE devices for the 2 Deg angle of camber case. The data showing the response for the original airfoil as a comparison helps to display the amount of difference between the clean airfoil and the VC LE option (see Figure 65).

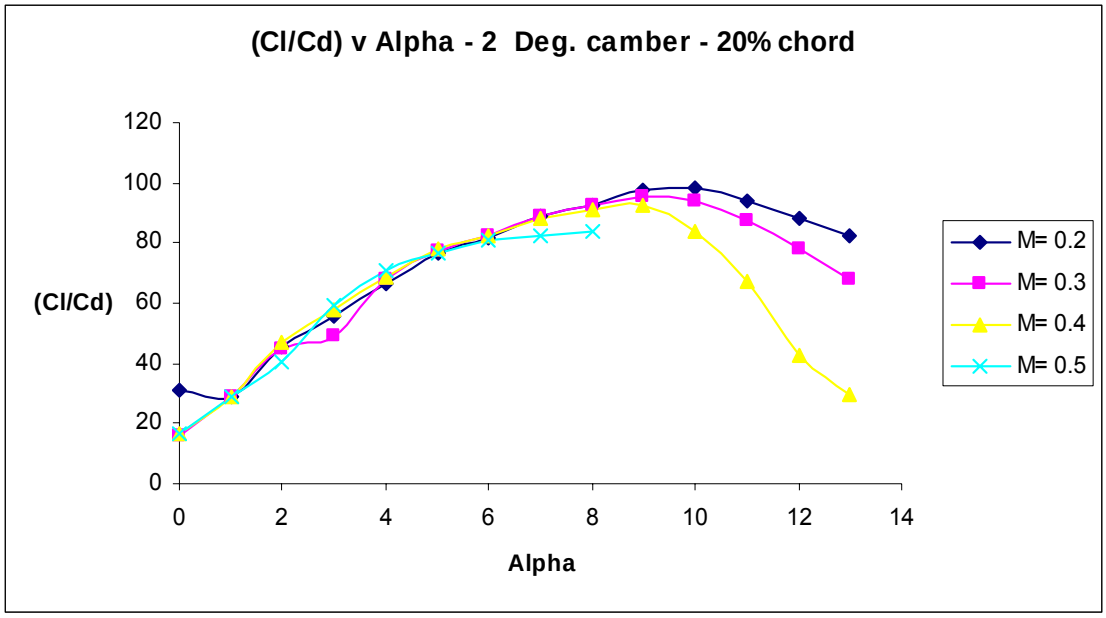


Figure 64 - Xfoil Results Cl/Cd 2 Deg Camber 20% chord[Lajux]

The limitations of the Xfoil software are such that it was not possible generate the equivalent set of results for a classical slat track option. Xfoil cannot simulate the flow around multiple components airfoil. It would have been interesting to see how such devices compare to the VC LE devices analysed in this thesis.

Also it is clear that for much higher angle of incidence the original airfoil is more efficient. This is mainly due to the extra drag generated by the camber of the LE at higher angle of attack. This is detrimental to the take off performance as it will slow down the aircraft; however this can be recognise as beneficial for the landing as it will reduce the overall approach speed.

Another thing to consider is the loss of validity of the results as M>0.4, as seen on Figure 64 the results associated with M=0.5 are limited to a lower angle of incidence. That is because Xfoil can not support this type of case. The effect of the viscosity start to appear and it show the limitation of this software. However, Figure 64 shows a clear trend for the L/D. The L/D ratio is decreasing for the same angle of attack as speed increases. For a limited range of alpha the L/D is similar regardless of the aircraft speed, that applies for the different camber angle and not to the original airfoil.

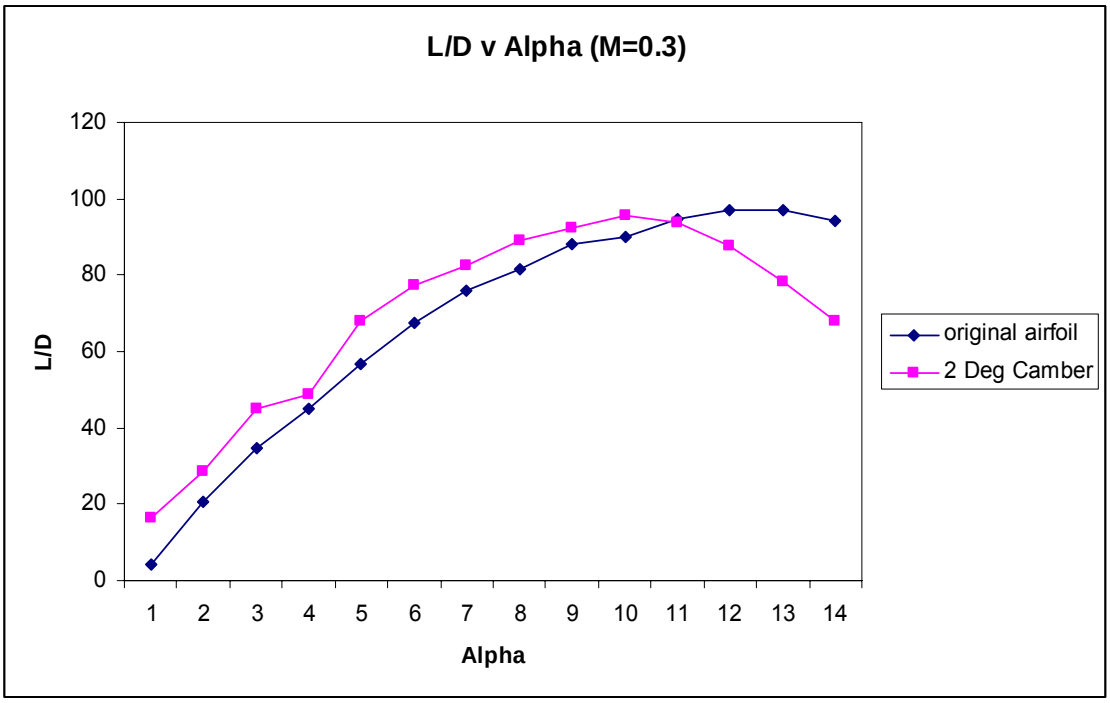


Figure 65 - Xfoil Results L/D 2 Deg Camber at M=0.3 [Lajux]

Also the VC LE design offer an increase of lift compared to the original airfoil, as seen on Figure 95 and Figure 96 in the appendices in page 176. It is important to see that the VC LE device allows a higher lift for the same angle of attack, this tie in with the usual behaviour of the more classical slat track concept. The high values for the overall L/D can be explained by the low accuracy on the drag result because of the viscosity effect on the Xfoil calculations. The two figures shows the overall behaviour of the L/D for the increase in speed and shows clear improvements with a 2 deg camber and an increasing speed. However, because a slat also includes a slot, this allows to re energising the airflow and in fact delays the stall. This is not possible to achieve such a result with a VC LE device as it will always keep a smooth continuity between the main airfoil and the LE device.

⇨ **High speed case**

The original airfoil was generated on GAMBITT to generate the grid for the CFD analysis (see Figure 66 in page 113). The set of points used to generate the airfoil profile were used from the data described before. They were then processed to create the airfoil profile, but this profile was also processed, but not modified, in order to accommodate the grid. The airfoil was scaled to have a specific chord dimension equal to 1 in order to simplify the internal calculation and provide output results already scaled. As the grid needs to be organised especially to investigate the effect of the different LE deflections it was necessary to separate the grid into several zones (or boxes). These were created depending on the shape of the airfoil sections created. These zones have a detailed arrangement of the grid for the different parts of the airfoil. It is also important to create the ideal node spacing as the nodes located further away from the airfoil will be more spaced out than the nodes near the airfoil. A refine grid close to the airfoil will increase the accuracy of the results and will help to show the boundary layer.

Once the different zones and the nodes spacing have been specified, it is possible to generate the grid for the overall model. It is understood that all the zones are connected together as they always share a common edge with the same node spacing. It is necessary to ensure continuity in the grid, to facilitate the flow calculation and to give a more accurate result.

The grid generated for the case study use different zones in order to detail the mesh as accurately as possible depending on the airflow along the airfoil. The different zones represent the different regions around the airfoil. The mesh is also large compare the airfoil chord in order to represent the natural condition for the air flow in the surrounding spaces. The different zones created for the case study generic mesh are described below (also see Figure 66 in page 113). Also it has to be noted that the grid is using a structured mesh.

The case study generic grid uses an average length more than to 20 times the airfoil chord both upstream and downstream (zone 1, 4 and 5) of the airfoil section for the “test area”. This allows the incoming flow to be as clean as possible in case of disturbance of the flow at the leading edge of the airfoil. It is also necessary to have the same amount of distance aft of the airfoil in order to see the turbulence further downstream of the airfoil. Reducing the distance for the grid forward and backward of the airfoil would in fact reduce the validity of the obtained results as they would be problems during the computation of the flow around the airfoil because the boundary of the mesh would still incorporate some of the changes in the flow characteristics (pressure, speed…). The mesh locate forward of the leading edge (zone 1) uses a circular organisation (Figure 68 in page 114), this allows to best mesh the curvature at the leading edge of the airfoil. This type of grid organisation allows best control for the meshing of the airfoil LE, the user can control both the number of nodes along the LE but also their position. This is extremely important for controlling the size of the cells along the LE profile it also help to control the transition with the zone 2 and 3. The LE section has been meshed finely to try to capture the boundary layer as accurately as possible (wall functions can also be used).

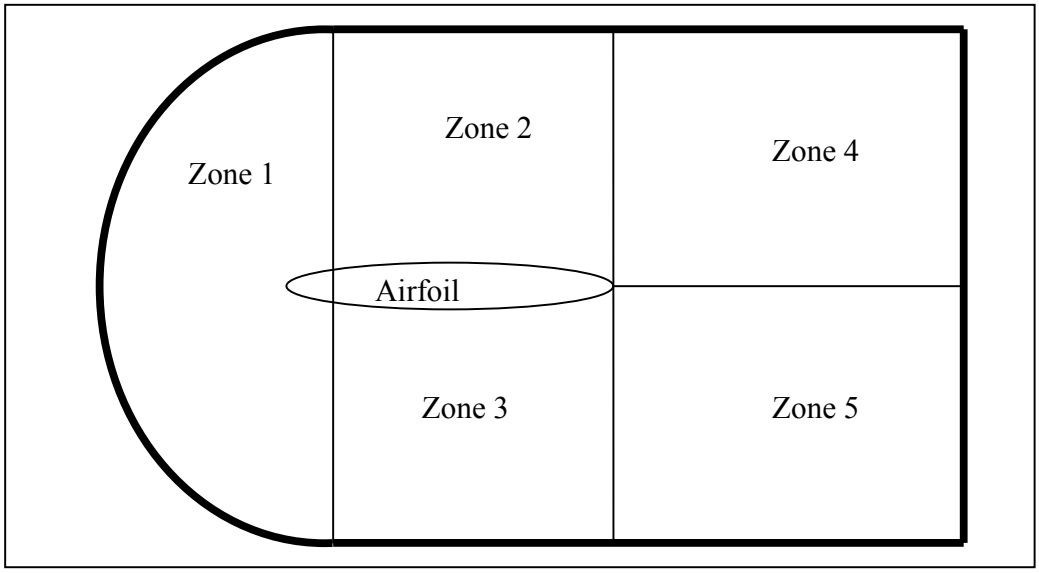


Figure 66 - Grid configuration for test case [Lajux]

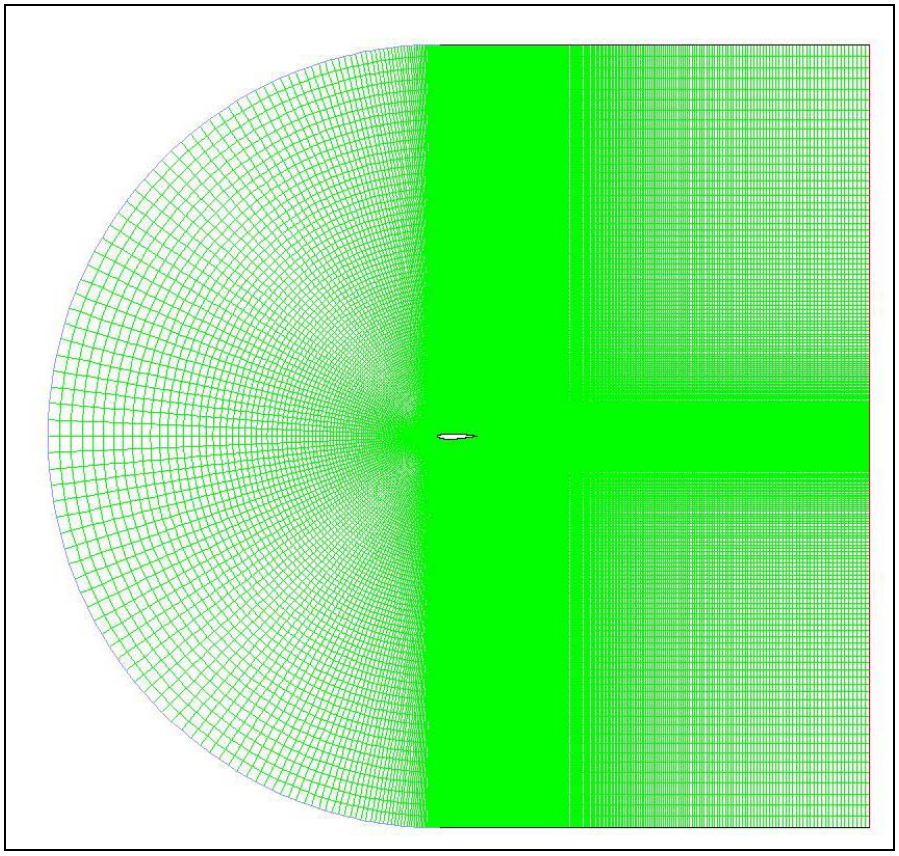


Figure 67 - Picture of overall grid [Lajux]

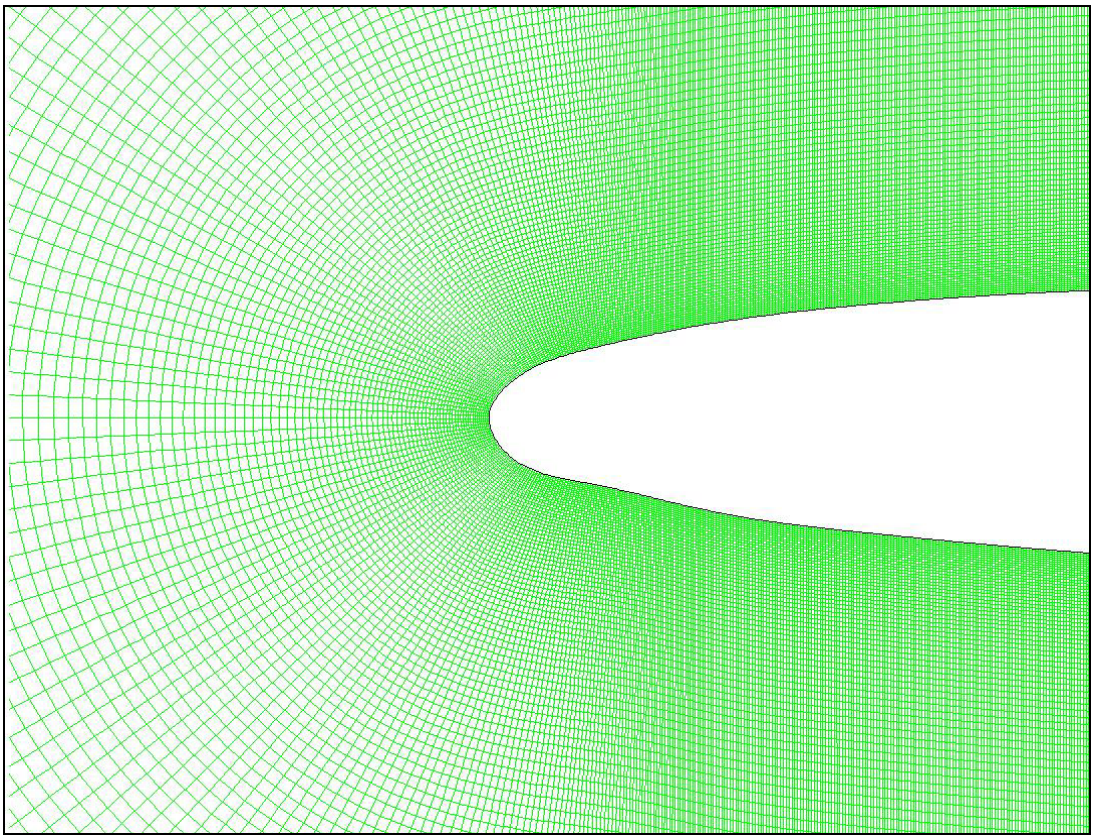


Figure 68 - LE mesh for case study [Lajux]

The mesh located backward of the airfoil is separated in two zones (zones 4 and 5) to get a more accurate, and flexible mesh to "capture the air flow behaviour after the airfoil. This is important for better representation of the drag as possible turbulence of the flow can occur much further downstream. Both zones 4 and 5 have a similar configuration as they are symmetric around a horizontal line from the rear of the airfoil to the end of the grid area. Also a gradient for the nodes spacing has been used both in the horizontal and vertical directions. The vertical direction has a refined mesh near the airfoil for better representation of the flow, and has nodes further away from each other as the nodes get closer to the outside edge of the mesh. Also on the horizontal direction the mesh is more accurate at the airfoil side. This way, with both horizontal and vertical directions using refined mesh it allows to capture the flow behaviour at the TE of the wing.

The mesh located above and below the airfoil (zone 2 and 3) is also important, as it will help to represent any changes in the flow behaviour along the airfoil. This will indeed help to see if there are any changes in speed of pressure along the airfoil, and especially if the flow becomes supersonic along the airfoil. Also the way these two zones are configured help the user to manage the link between the LE and TE mesh. Zone 2 and 3 in fact control the vertical mesh organisation (node spacing and numbers). Figure 68 shows details of the LE junction with these 2 zones, it is possible to see there the jointt between the LE meshing zone (zone1) and the zones 2 and 3. The link between the zones

is very important; it is required to manage well this link. The gradient in cell size between zones can be detrimental to the final results if not managed well. In fact, by changing cell area too rapidly between two neighbouring cells will give bad results due to the calculation possibly going wrong between these cells. This would represent a loss of accuracy of the results.

Once the grid has been created around the airfoil it is possible to define the characteristic (or physical properties) of each one of the grid edges. This is to represent the displacement of the flow as well as simulating the boundary conditions and making the virtual model as close as possible of the real conditions. After all this has been done it is possible to send the overall grid and the boundary conditions (as well as the airfoil) into the FLUENT software to run the simulation and get the results.

At this point of the simulation it is important to represent the virtual model and the flow properties (including the boundary conditions) as accurately as possible, and also to make sure that the flow speed is the correct one. The flow speed has to be calculated relating to the dimension of the airfoil used within the CFD software. As the airfoil has been reduced in size to have a chord equal to 1, it is necessary to adapt the flow speed to keep the same Reynolds number as the one for the real condition (equivalent to full chord length).

When all the parameters to run the simulation have been set up and the model is fully defined it is possible to launch the CFD case and get the results a few hours later.

The user had to verify that the simulation had converged to a solution through the observation of the value for the residuals which will show the difference of results between each of the iterations. As the process starts, the value of the residual will normally change by a large amount between each of the iterations, this is because the airflow is still unsteady as it goes through the test section area. Indeed, the residual,s when converging demonstrate that the system has reached a more or less steady solution as there is only little difference for the characteristics of each cell between 2 iterations. Then the user can be confident that the solution will be valid, or at least that the solution has reached a steady behaviour. An example of the residuals converging towards a steady solution can be seen in Figure 69 (page 116).
This graph shows that the residuals after around 5000 iterations do not change too much, indeed they change by an order of magnitude of 10^{-8} for most of the residuals and only 10^{-3} for the continuity residual. That shows clearly that the amount of change during the last 5000 iterations is minimal and that the result is expected to have reached a steady state.

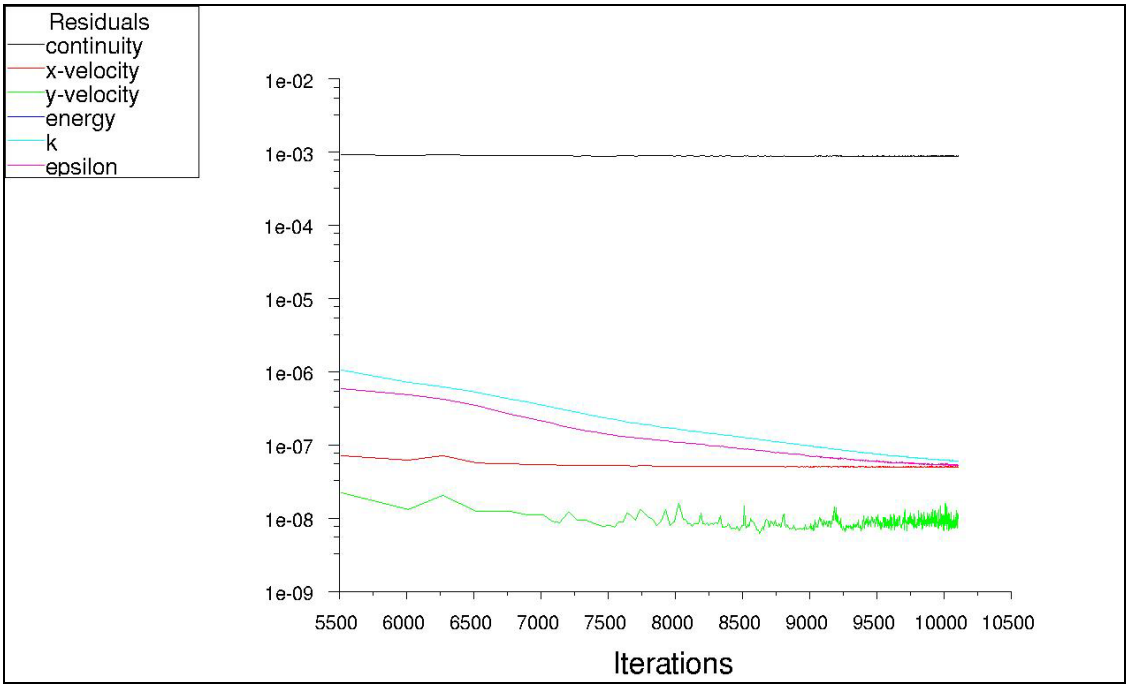


Figure 69 - Example of residual convergence [Lajux]

Once the CFD simulation is finished for each case, the user can get the output results for the aerodynamic performances of the airfoil. These outputs can be the pressure coefficient distribution around the airfoil, the air velocity within the grid or the density of the air… Many outputs can be generated to obtain a description of what is happening around the airfoil.

The Figure 70 and Figure 71 show some of the results obtained during this analysis. These figures are used to demonstrate the L/D compare to a change of angle of attack or compare the L/D for different settings. All these cases are using the same speed, specified for the high speed case (M=0.8) in order to represent the cruise conditions. More data on the high speed case can be found in Appendices D (see page 177).

It is clear that the using the VC LE even with a small deflection (1Deg) can provide some interesting results as it increase slightly the L/D ratio (see Figure 70), therefore it generate more lift for the same amount of drag. This would be indeed beneficial as the aircraft fuel burn rate through speed reduction to obtain the same amount of lift with a reduce drag compare to the original airfoil.

However, it also appears that the change in camber does improve the L/D, but as the camber increases this L/D offset benefit is reducing, as it is expected that once again the higher camber will generate more drag.

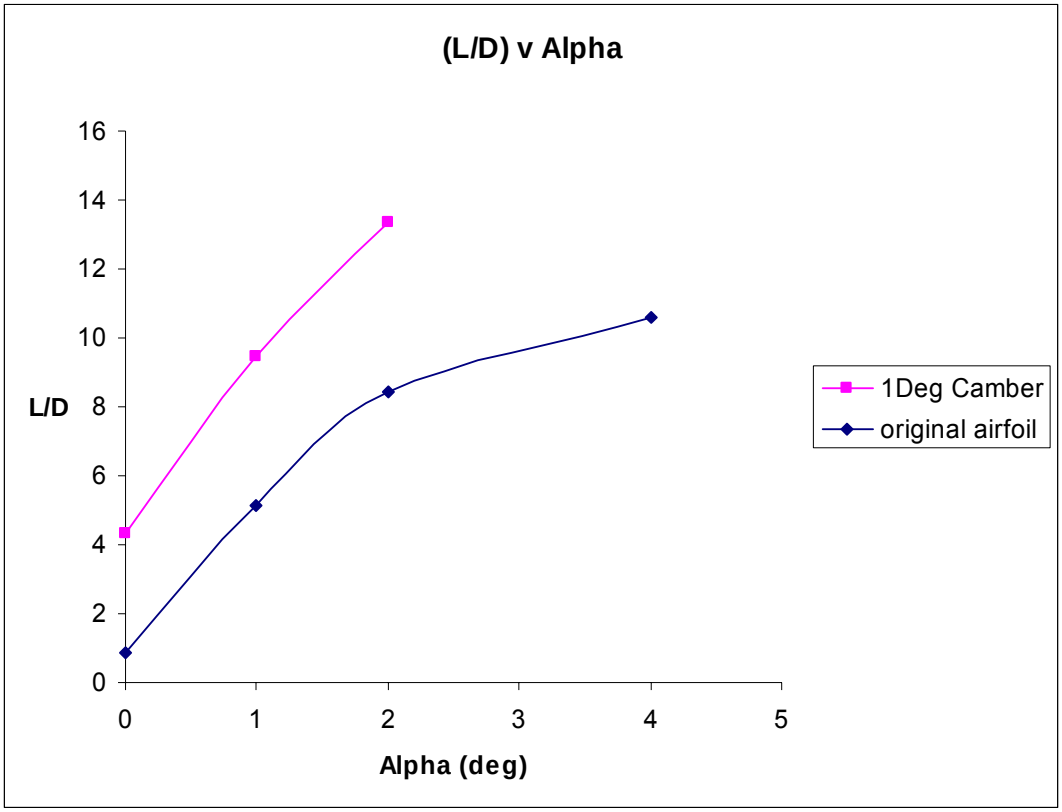


Figure 70 – Relative L/D high speed case for L/D v. Alpha changes [Lajux]

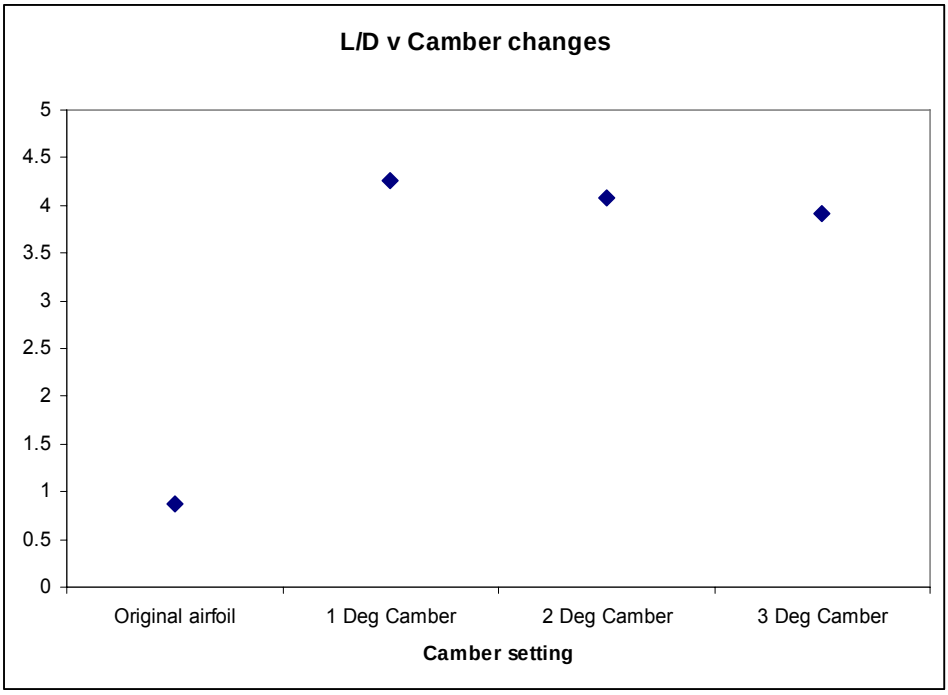


Figure 71 - L/D high speed case for L/D v camber settings [Lajux]

5.6. Structural and Mechanical design

In order to design the ATRA LE device deployment mechanism for a VC application, it is necessary to go through a detailed process. This process has been fully defined in paragraph 3.5.1 of the design methodology, with a precise explanation for each step. For this part of the methodology it is intended to design the deployment mechanism, but it is also expected to create a structural model of the mechanism using CAD and FEA software. This model will be able to support the different loading cases which are applied to the wing during the different flight conditions. All of this will be fully explained in the following paragraphs. Also results will be shown when it was possible to achieve a complete result. If it was not possible to reach a final solution the process is explained for getting a possible solution in future work.

5.6.1 Kinematic design

As explained in Chapter 3 the design of the mechanism is done in two steps, first to generate a 2D optimised kinematics chain model, and then to produce the whole 3D model including the inboard and outboard section.
First of all it is only possible to start with a 2D design.

- **Mechanism synthesis (2D):**

The 2D design of the LE device starts with the knowledge of the exact generic airfoil profile definition. This includes the knowledge of the exact position of the front spar, the available space to fit the mechanism and the possible position for the fixing points.

This first step is the most important as it defines the "working envelope" in which the designer will be able to package all the different elements or components for the LE, as well as the (possible) position of the front spar support points. It is also necessary to generate the different camber settings to model the deployment trajectory (see Figure 72).

It has to be mentioned that at this stage the designer does not use any detailed design for the flap component itself. This is only a synthesis, and then follows the optimisation of the mechanism and not the flap design. The detailed design for such a component will be studied at a later stage.

With the generic airfoil profile and its different configurations, it is possible to generate the ideal path for the deployment of the LE device for the different camber settings. However this deployment path could be defined in different places of the LE slat component profile, as long as it is located at the front of the front spar. But that position could have a noticeable influence on the resulting kinematics chain and mechanism geometry.

Once that is known, it is possible to start applying all of the data on the CAD software to generate the first model for the mechanism synthesis. At this stage it is possible to decide the exact position for the deployment trajectory as well as the possible position of the fixing points for the mechanism. For the synthesis there is no need to include the front spar as no element clash can be detected. This will be taken into account at a later stage.

Even if the front spar is not designed it is important to know where it will be located as the location and range of the different points (and components) should not be after the front spar.

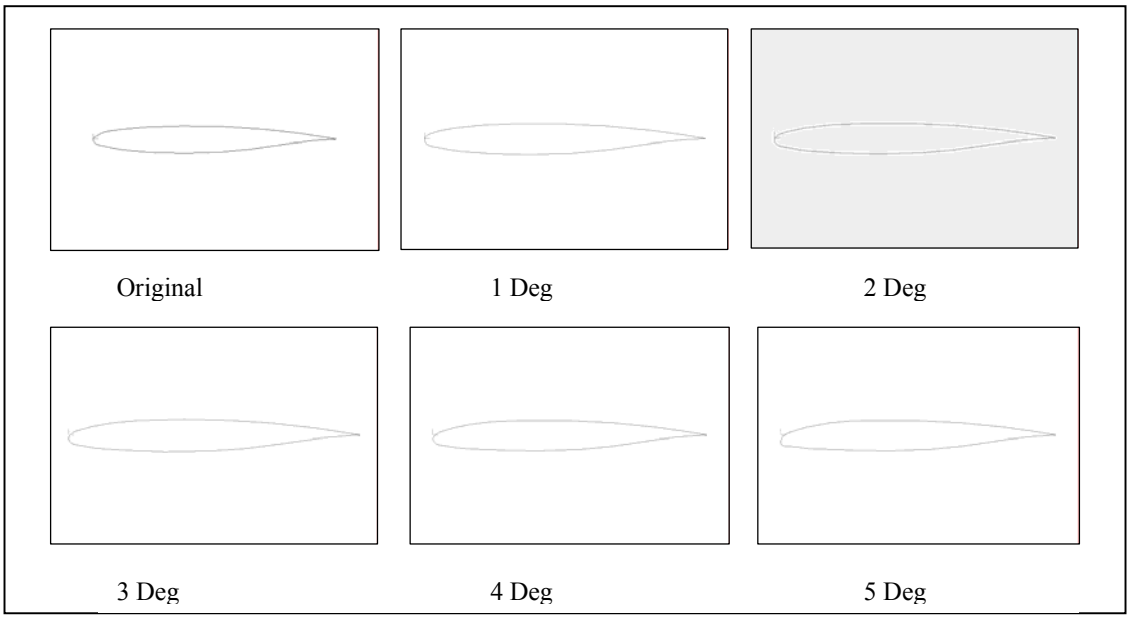


Figure 72 - Different camber profile [Lajux]

With this data it is possible to launch the SYNAMEC software to find a possible kinematics chain which will comply with the given objective function (deployment trajectory), and position of the different fixing points.

The SYNAMEC software which was developed during a European Research project [29], for which Cranfield University was a member of the consortium, has been made do develop new mechanism in relation to a given deployment trajectory and a set of fixed points. From there the software generates possible kinematic chain which also contains the kinematic joints (hinges and others). This kinematic chain is normally giving an approximate answer to the user request, for this reason it is necessary to optimise the generated model on a second stage. This optimisation will be done for the case study in the next paragraphs.

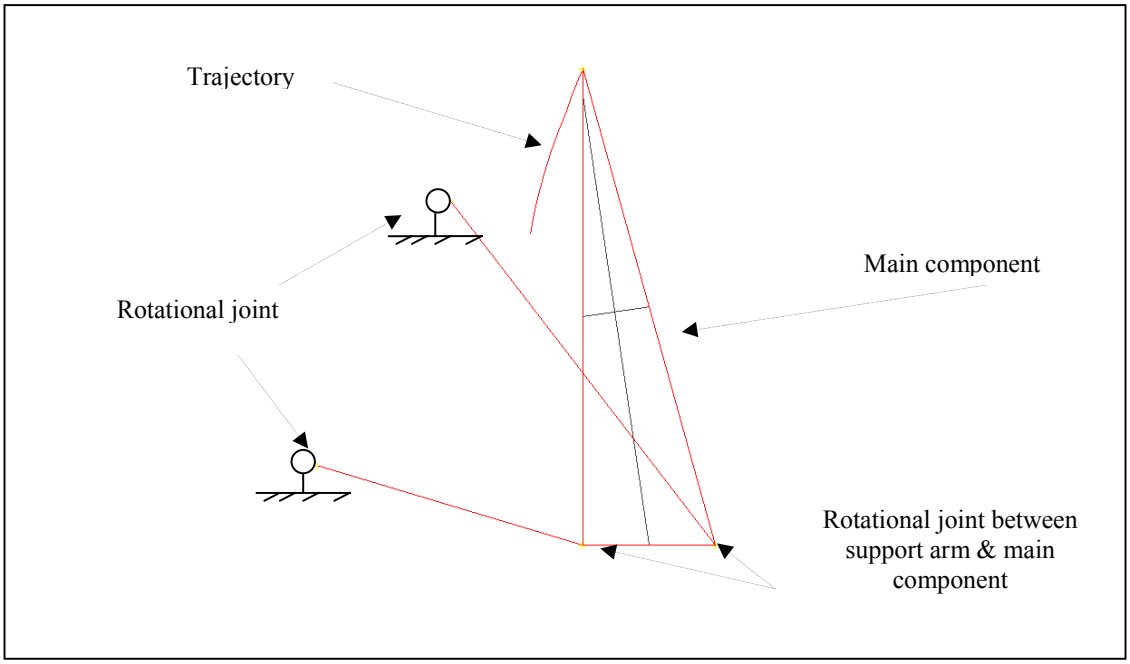


Figure 73 - Mechanism description [Lajux]

The above figure describes the mechanism generated to deploy the LE. It also describes what type of kinematic chain has been created. At this point the LE device is not seen; a picture of the mechanism in relation to the front spar and LE device skin can be seen in the next figure (Figure 74).

The produced kinematics chain (Figure 73) is made of a solid part (with 3 attachments) and two bars which link the different fixing points of the front spar support to the created. The two bars fixing points represent the points which will support the mechanism and which is linked to the front spar. Also, the triangle component makes the link between the two bars and the slat component. The kinematics jointts are already defined by the software so it is then only necessary to work on the detailed optimisation of the mechanism, and the resultant deployment trajectory.

This kinematic chain is quite simple and does not require many rollers or bearings, it would only be necessary to have them at the junction of the triangular component with the deployment arms and also the fixing points. By having a simple mechanism the cost for the manufacturing and maintenance will be kept to a minimum. Also it will be important to consider how this new mechanism will be actuated and where these actuators could be fixed. This will be discussed in the following paragraphs.

For these reason it was chosen to select this kinematic chain, other deployment mechanism such as a track and roller arrangement have been considered, however, this require an intrusion in the fuel tank located in the wing box. This new kinematic chain will provide a design solution with no intrusion within the wing box, which is an

important point as it will reduce the time to design the fuel tank, the front spar and look at all the sealing problems linked with intruding within the fuel tank in wings.

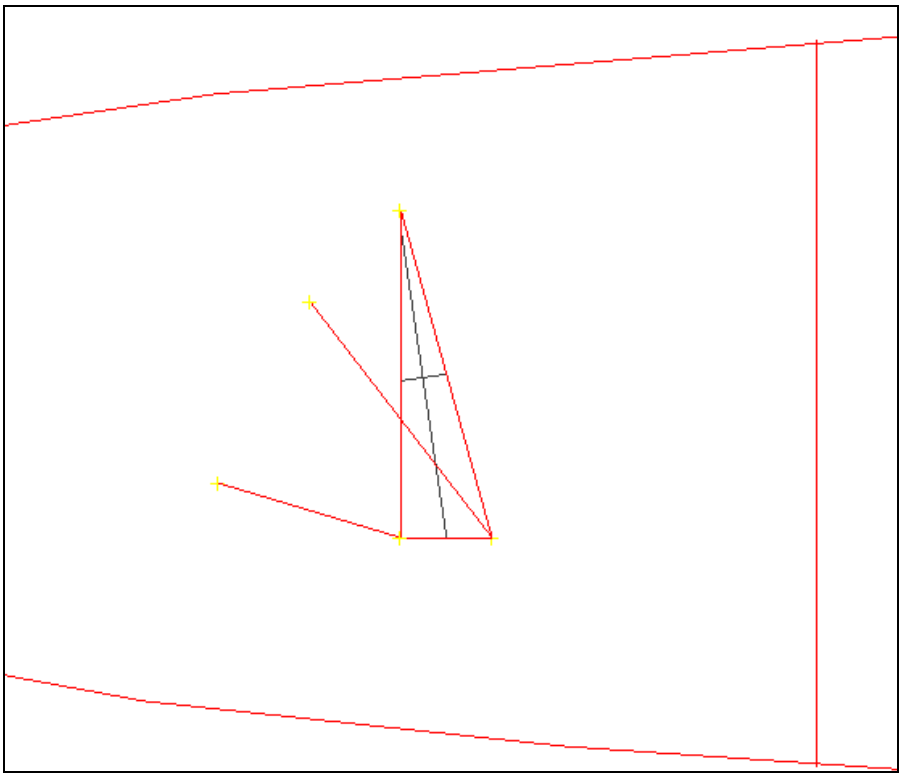


Figure 74 - LE mechanism with airfoil and front spar [Lajux]

⇨ **Mechanism optimisation:**

The optimisation is carried out by making a parameterised model using features from the kinematic chain described in the preceding paragraphs. This is done by using the coordinates of the fixing points as parameters. This gives the opportunity to investigate the effect of the different positions of these points compared to the overall solution. Figure 75 shows the optimisation process using Boss-Quattro [60], this figures shows the link between the CAD model, the kinematic analysis and the optimisation

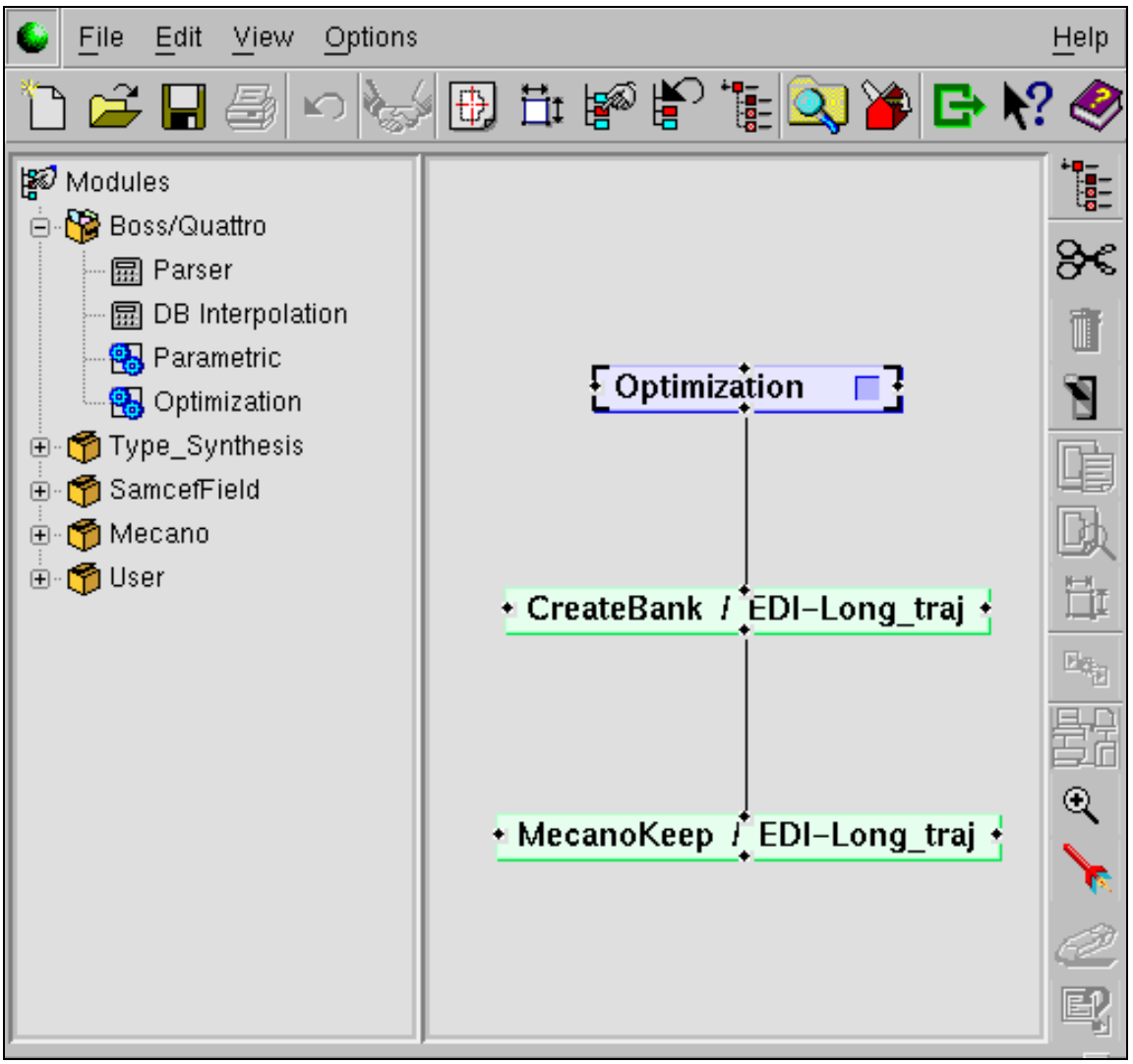


Figure 75 - Optimisation structure [Lajux]

To find the possible range of variations for the different points it is necessary to give the constraints for these parameters. Usually constraints will be the exact position of the front spar, the estimated position of any rib on which the mechanism will apply the motion, and also the location of the slat skin. All these constraints will give the overall design envelope to fit the mechanism. Having a wide range of variation makes it easier for the optimisation tool to find a suitable solution.

In order to achieve a good result for this optimisation the genetic algorythm capabilities of Boss Quattro has been used, this allows the user to investigate a large design space using a large variation (and non linear) of the design variables to achieve a better convergance of the objective. All the design variable will change during the optimisation process related to the genetic algorithm method, which offer the possibility to change the rate of deviation of the parameters between iterations. This indeed helps to investigate different possible solution for the kinematic chain.

Figure 75 shows the overall process for the Boss-Quattro optimisation, the first step is the connection with the kinematic model provided during the kinematic synthesis stage. This is followed by the the creation of the "bank" or data base of the design variable (with the range of variation), the constraints and the objective definition. These different parameters are described and explained in the following paragraphs.

To do such an optimisation, it is necessary to define the exact parameters and their possible variation. The following paragraph gives each parameter location and also gives the range of variation.

The variables listed below are the coordinates for the two attachment points and the three corners of the triangular part.

Initial position:

Pt 1

Initial position:	X=	550	mm		Y=	50	mm
Range of variation:	X=+/-	100	mm		Y=+/-	100	mm

Pt 2

Initial position:	X=	500	mm		Y=	-50	mm
Range of variation:	X=+/-	100	mm		Y=+/-	100	mm

It is also possible to use some of the points of the triangle component.

Pt Triangle 1 (bottom left):

Initial position:	X=	600	mm		Y=	-80	mm
Range of variation:	X=+/-	50	mm		Y=+/-	50	mm

Pt Triangle 2 (bottom right):

Initial position:	X=	650	mm		Y=	-80	mm
Range of variation:	X=+/-	50	mm		Y=+/-	50	mm

Using these parameters and their respective variation, it is possible to optimise the solution to get a more accurate design. This design gives the same type of kinematics chain but with new dimensions and locations of the different components which will help to further improve the accuracy of the deployment trajectory. Following the optimisation process it is possible to get an optimum solution for the deployment of the mechanism compared to the ideal deployment trajectory of the LE device.

In fact, the fixing points are the ones expected to have some surrounding empty space. Also the points used to fix the solid part to the LE slat can have different positions as long as it is not too close the slat skin. The range of variation for the different parameters can only vary depending on the available space. For this case there is 20% of the overall airfoil chord available, also the nose of the LE device is quite narrow so this will have to be taken into account as a limitation of the available space.

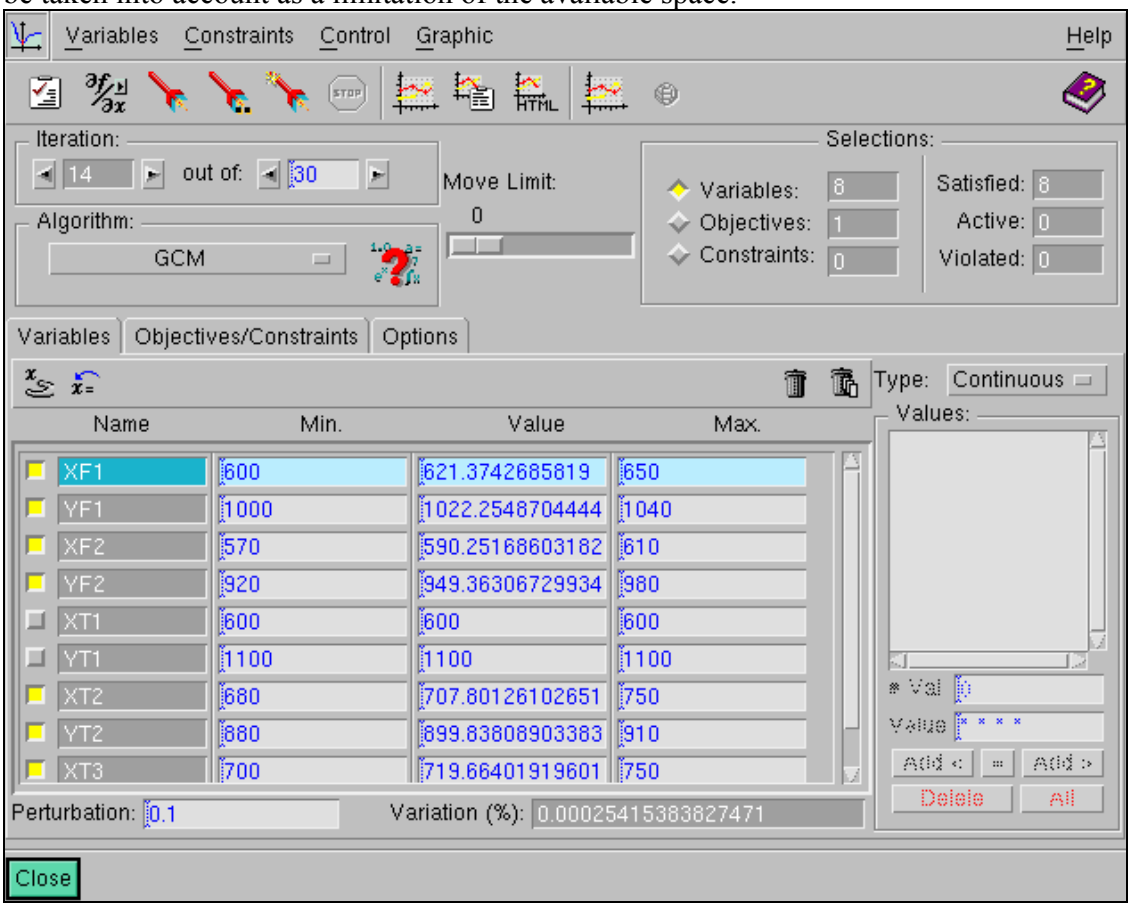


Figure 76 - Optimisation parameters [Lajux]

From this point it is possible to run the optimisation and possibly use the variables and change their values to see if it is possible to obtain different results (see Figure 76). The variable will change value for each iteration during the optimisation process. The optimisation uses a genetic algorithm to refine the initial solution in order to comply as accurately as possible as the objective. Generally, during the first iterations the optimisation software modifies largely the different variables to see which one will have a larger influence on the output result. This way it automatically drive the solution towards in improve configuration.

For this case study, as stated earlier in this chapter, the objective to optimise is the deployment trajectory , this is in fact a way to improve the kinematic of the LE device in order to match as closely as possible the shape described by the aerodynamic study. In fact, the design variables (also called parameters) are given a range of variation, defined by the user, to define the possible optimum location of the rotating joint within the LE

enveloppe. The initial values for the location of each design variable is given by the kinematic synthesis results provided in the precedent pages. The constraints are based on the range of actuation defined by the user (that is depending in the radial motion range of the actuator) and the space enveloppe available. However, the space envelope constraints is undirectly applied to the optimisation process by the description of the range of change in the location of the rotating joints. The author used only values for the variation of the rotating joint which guaranteed all the point to be within the LE space envelope. The actuator range was chosen to be 30 Deg as a standard value for the deployment of the LE device.

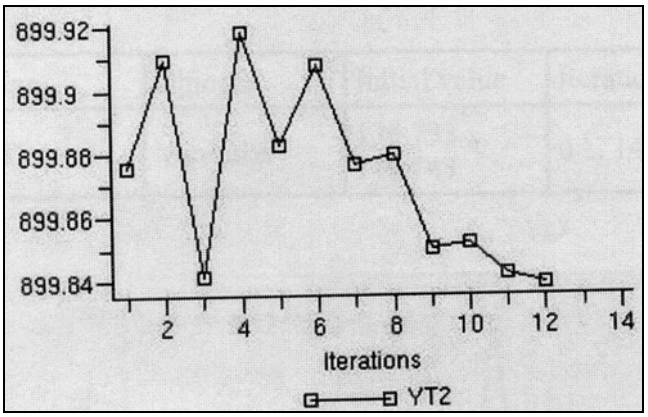


Figure 77 - Dimension optimisation graph [Lajux]

Figure 77 display the evolution of the parameter YT2, it shows the changes of the value of the YT2 parameter which is the coordinate along the y axis of the point 2 (stated in mm on the vertical axis). That represent the behaviour of one parameter (or design variable) during the optimisation of the objective.

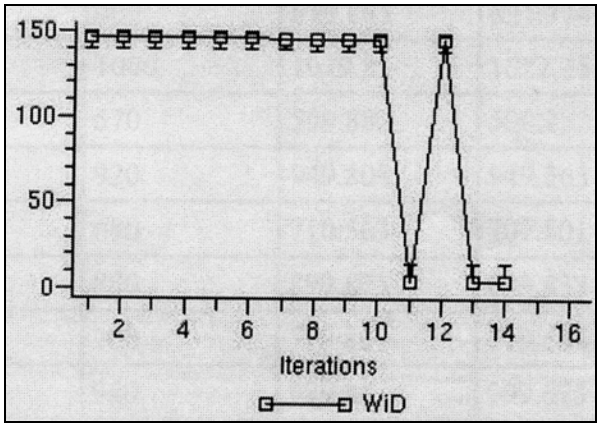


Figure 78 - Objective optimisation [Lajux]

Figure 78 display the evolution of the objective optimisation, which is the distance between the ideal deployment trajectory and the current deployment trajectory at each step or iteration of the optimisation process. The vertical axis displays the value in mm of the distance between these two trajectories.

Due to the small amount of space available for the different points to be located, it is quite difficult to optimise by large margin the kinematic chain generated by the SYNAMEC tool.

The Figure 77 describes the evolution of one of the point coordinates (along one direction only) compared to the iterations. It is clear that the optimisation process reach an optimum point as small changes in the position of this point happen during the last 3 iterations. The Figure 78 shows the results of an optimisation run which took 14 iterations to complete the objective. After two iterations which give the same results the optimisation stops, this is also linked to the optimised value of the parameters which are converging towards a stable value for an optimise objective.

However, regarding the results obtain in this optimisation; it is clear that the optimisation process can be unstable. In fact, the large changes of range for the objective dimension value means that small changes in the variable will have a large effect on the objectives, it also means that the mechanism has to respect the position of the different points very accurately. Only a small inaccuracy on the positioning of one point compared to the other could change the outcome of the deployment trajectory. That could be critical during the assembly process as controlling the tolerance between different elements will be difficult.

The sensitivity of the kinematic chain depends on the initial position of the design variables , and so it is important to have the chance to use the kinematic chain synthesis tool in order to generate kinematic chain with have already a generally “good behaviour”.This is shown by the different figures in the precedent pages, as little change of the design variable influence greatly the optimisation objective.

Also the the space enveloppe constraint was given by the limitation on the possible range of changes of the rotation joint location, this proves critical as it would be possible to get better and more acurate solution if the rotating joint location could vary by a glarge margin. This would generate poor design solution as the overall kinematic chain would be outside of the airfoil profile.

The precedent comments show the importance of the user uderstanding of the constraints to design a LE deployment mechanism and so put a great emphasis on the user of the optimisation software. The fact that the objective is so sensitive compare with the variations of the design variable also means that the initially produce kinematic chain was close a perfect match for the ideal trajectory deployment.

⇨ 2D Mechanism design and expansion:

Once an optimum solution for the generic 2D case has been defined, it is possible to follow the same process to design, or "scale", the different mechanism along the wing. This 2D model will be the starting point for implementation of the 3D cases.
However, if it is possible, it is important to try to keep the motion cylindrical to keep the design and manufacturing cost down. This would create a simpler design solution to extend the LE device, which in fact will use the same mechanism along the wing span to deploy the different panels. That would also reduce the maintenance cost at a later stage as most of the components will be similar with no variation of size and shape.

⇨ 3D Design

It is possible to generate a full 3D model of the overall LE systems by adding up the different mechanisms. This process is the same and does not depend on the 3D deployment motion (conical or cylindrical). This is obviously taking into account the geometry and location of the inboard and outboard section, as this has implications on the overall design. A conical deployment means that the mechanism will need to be scaled along the span of the wing. To design the full 3D mechanism system it is necessary to define the exact location and dimension of the sections, and each panel in these sections.

Location and dimension of the inboard and outboard sections:

Once the mechanism geometry and the position of the different points have been fully defined, it is possible to study the structural design of such a mechanism. However, it is important to look at the initial mechanism solution. Figure 79 shows the initial 3D solution for the mechanism produced during the synthesis and optimisation.

The driving mechanism was not discussed so far, only the kinematic chain to deploy the LE devices was discussed. There are different possibilities for the actuation system, as described in the technology review in Chapter 2.

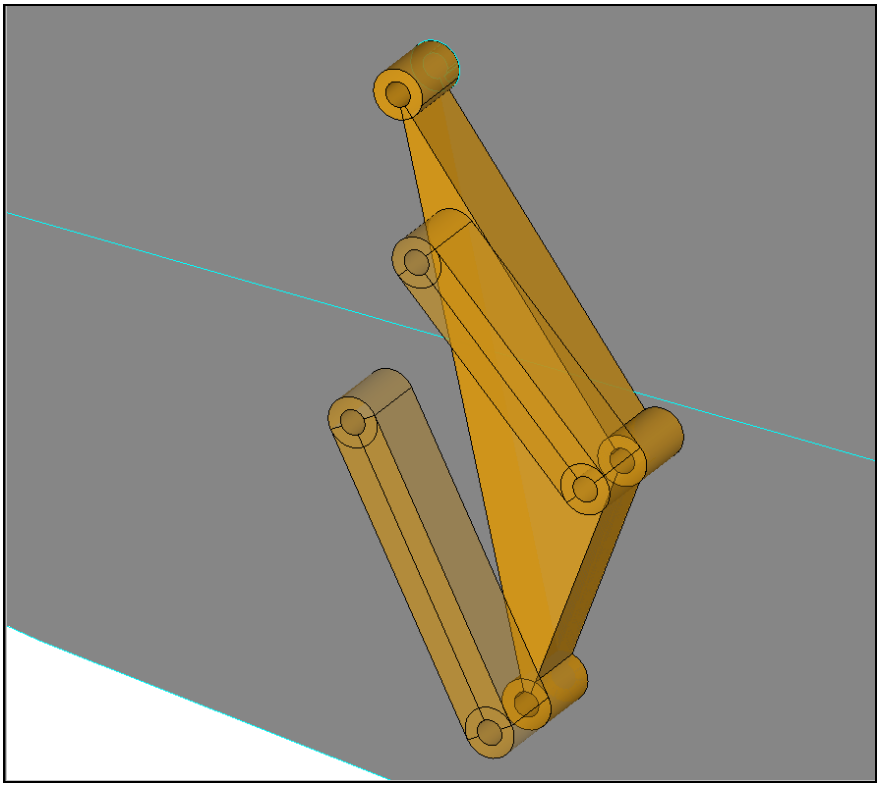


Figure 79 - 3D picture of Initial mechanism solution [Lajux]

One of the possibilities would be to have a rotary actuator located at the fixing point of one of the deployment arms, leading the motion of the triangular part. This actuator would be electric and would only require an electric link to the flight control computer system. That would be an ideal option as each actuator would be driven at appropriate speeds and this would be the perfect solution in the case of conical deployment of the slat. Another important point would be that the pilot could have more control over the exact deployment speed and position of the slats. This is also an ideal solution for the VC camber option as the pilot could refine the deployment of the slat at different camber amber in cruise condition.

However, electrical actuators are well known for the lack of torque and this would especially apply to the loading to counter act aerodynamic loads in cruise conditions. An electrical rotary actuator would in fact have problems to deploy the slat, but more important would surely have more problems to keep the slat in a fixed position and counter act the aerodynamic forces.

Another solution would be to have a sliding, or linear, hydraulic actuator (piston) pushing the rotating arms around the fixing points. This way the actuator would be fixed on the front spar and would have a direct contact with the rotating arm. That would provide a direct load path back into the front spar which is ideal instead of loading the actuator. The only problems comes from the control of the deployment of the linear actuator, but using state of the art control system it is expected that the driving system could be control

following a small displacement step. This way the pilot would still have full control over the deployment of the slat, even in cruise condition.

In order to complete the design of the VC LE device it is necessary to investigate how this structure will joint and interact with the surrounding structures (upper and lower wing covers). The following figures, Figure 80, Figure 81, Figure 82, and Figure 83 show the proposed solutions for the junction of the LE device and the covers. Figure 80 shows the overview of the VC LE device, and Figure 81 show a zoom in view on the joint with the upper skin, the Figure 82 and Figure 83 show a zoom in of the joint with the lower skin.

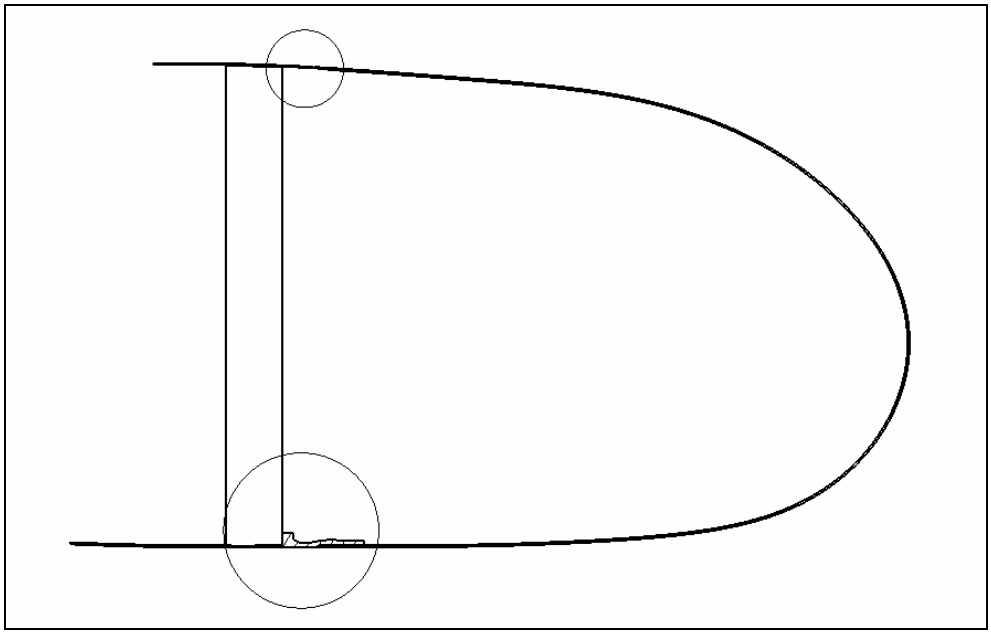


Figure 80 - Overview of the LE to skin joints [Lajux]

There is only one design solution proposed for the joint of the LE to the top skin. This is because it is expected that the LE device will slide along the riblets positioned along the front spar, and, use them and the chamfered end of the upper skin as a support. The Figure 81 shows the example for this concept. The top skin, located on this sketch on the left side is slightly chamfered to welcome the top edge of the LE device, which itself is chamfered on a parallel direction. Using this concept it is clear that the LE device will slide along the top of skin as it gets deployed. Also this concept makes it easy to design the interface geometry, as the deployment trajectory is fully known.Rubber could also be added to the tip of the LE device top edge to guarantee smooth continuity during deployment.

Also it has to be noted that the upper skin should not lift under the effect of suction. This lifting or displacement will be analysed in the structural study paragraph. It is planned that structure of the LE device, by means of ribs and stringers will control this deflection and that the skin will not by submitted to large displacement due to the loads.

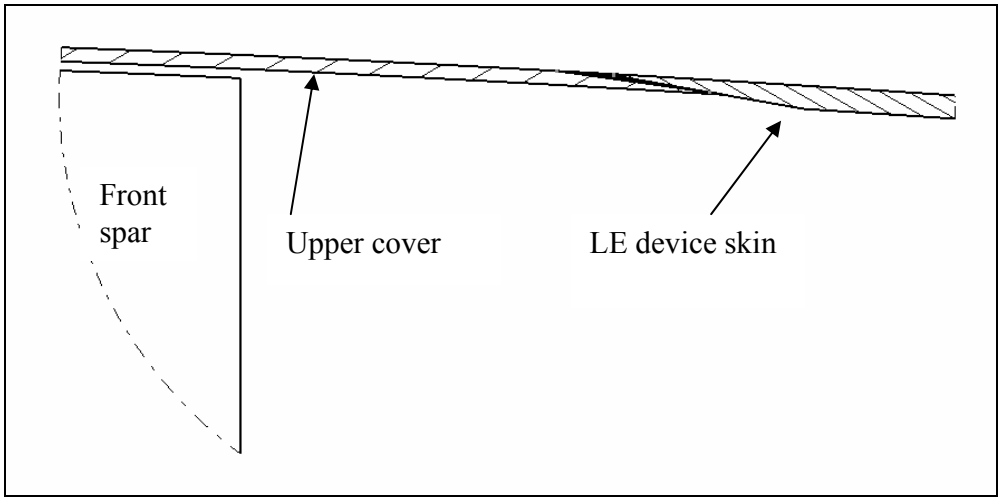


Figure 81 - Top skin joint [Lajux]

For the bottom skin there are two proposed concepts, as this is a different case. The LE device when deployed will have its bottom edge going slightly down compared to its original cruise condition. That means that it will be required to have something filling the gap when the LE is deployed.

The first design solution proposed is to use a rubber part which is in compression in the initial position that way when the LE device will deploy the rubber part will expand to fill in the gap as it will want to retrieve its natural configuration. This solution also guarantee to have a fully flexible skin.

Also it is planned to have an internal spring metal sheet to keep the rubber part in contact with the bottom edge of the LE device.

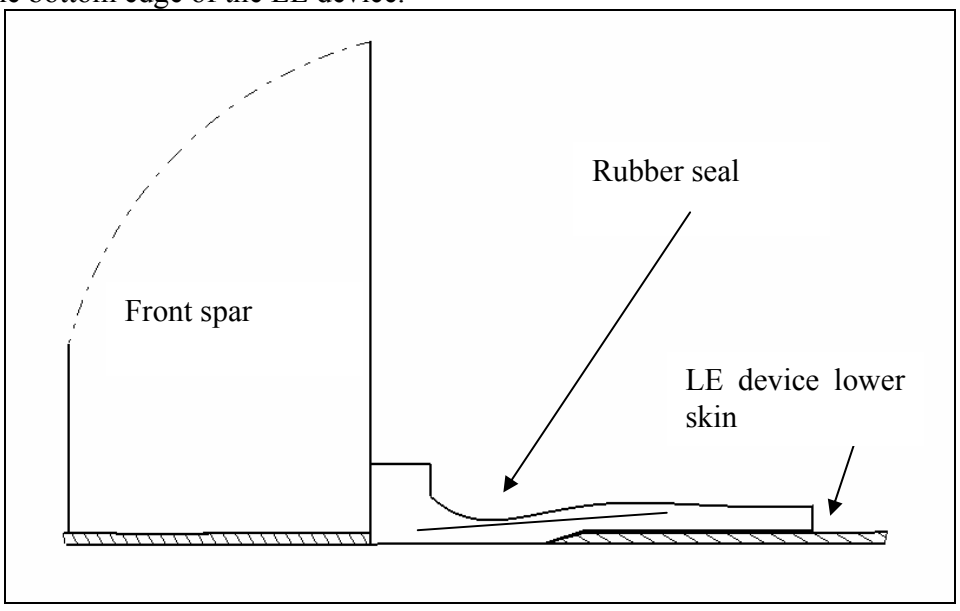


Figure 82 - Lower skin joint concept 1 [Lajux]

The second design solution proposed for the joint of the bottom skin and the LE device is to use a plate rotating around a possible hinge, this plate will be pushed with a spring attached on a bracket on the front spar. This plate is also chamfered to offer a smooth surface when the LE device is deployed and to keep a regular contact between the two components. This way there is always a smooth airfoil profile.

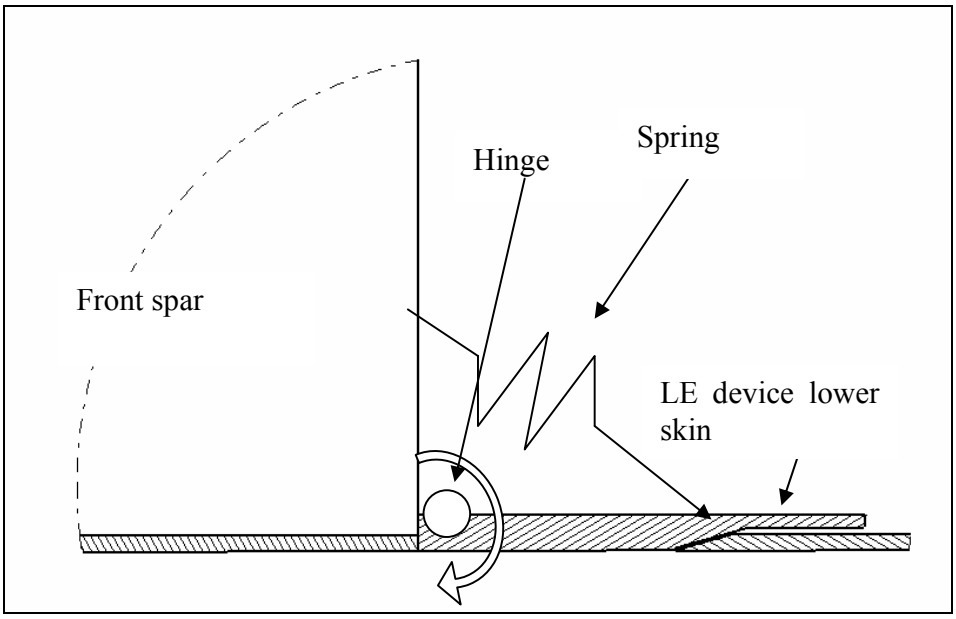


Figure 83 - Lower skin joint concept 2 [Lajux]

The concept 2 is slightly more complicated than concept 1, and it will also require more parts and a longer assembly time, which increase the overall cost of this solution. However, this concept would guarantee a continuous contact between the two surfaces, instead of the rubber solution which might not guarantee to insure a continuous contact. This is because the load applied on the rubber might be greater than the load the rubber applies on the edge to retrieve is original position.

For this reason the concept 2 should be used for this case study and it could also be investigated further.

Now the mechanism and the overall geometry is finally defined it is possible to pass to the next step of the methodology, which is the structural analysis of the different components to see if they comply with the applied loads and also to see if it is possible to optimise them and generate some weight saving.

5.6.2 Structural design

The structural design of the deployment mechanism has to cover the study of the sizing of the different elements (to keep the mass down) and also to see if the mechanism will resist the different loading conditions. These have to be optimised to provide a result which is the lightest possible, whilst also complying with the critical loading.
This structural design will be done in two steps, first the sizing of the different components will be done using slender beams, and secondly a more detailed 3D FEA study of the components can be done.

- ⇨ **Initial design (slender beam)**

The first step in order to size the components of the mechanism is to represent each of them as a single beam element. It is then possible to give these "beams" some generic properties. This covers the cross section definition as well as the physical properties of the material used for each component.

Once the different properties have been defined it is necessary to generate a mesh for each component, in order to apply the loads to the overall mechanism. For the initial design only the 2D models of each mechanism are considered.

The loads are obtained from the aerodynamic study. It is necessary to take the highest loading case, also called critical loading or ultimate loading case, to be sure that the chosen design will resist this loading situation. At this point it is also important to refer to the different set of regulations regarding the definition of the critical loading case. The regulation can give a defined value for a safety factor to add to the critical loading case. For this reason it was decided to use the landing case[21] under a 2.5G, as this is the worst loading case for sizing the LE mechanism.

To obtain the loading on the wing it is necessary to know the pressure distribution (spanwise and chordwise) along the wing. From that it is possible to derive the load applied to the wing using the formula mentioned in Equation 3-9 (see page 70).

When each of the 2D mechanisms along the wing have been analysed it is possible to do a 3D study of the overall mechanism system to see how it will react to the loading all together.

This of study was not done for the case study but the overall process has been explained for use in future projects.

⇨ **Detailed design (3d components FEA)**

The detailed design phase will in fact define the final size and shape of each component. This is normally done using 3D models of each component and a FEA study of these components. The 3D shape of each element can then be improved to reduce the component mass and also to improve the resistance to crack and fatigue.
This detailed design will help to fully define each component geometry and physical properties.

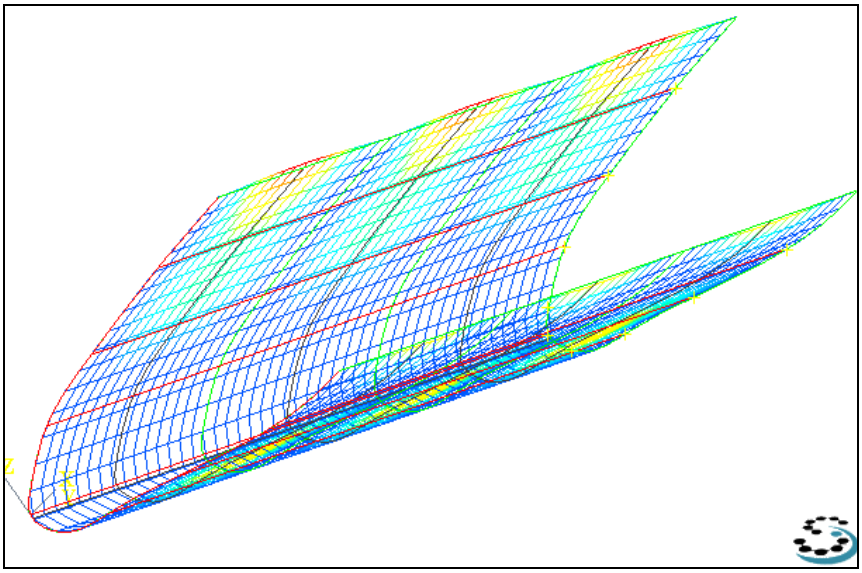


Figure 84 - Nodal displacement for slat (max. load & cruise profile) [Lajux]

The first step for the detail design is to get the initial 3D model of the different components or at least the slat skin and the components for the mechanism deployment. With these 3D models it is possible to create the adapted mesh for each component and then to fully define the model.

First of all it is possible to define the components physical properties and materials then it is possible to do the mesh for each components, this is taking into account the different ways to represent each components within the FEA software in order to obtain a quick and accurate solution. Also it is important to define the physical properties of the different components and to fully constrain the model in order to represent the real conditions applied to the LE device. For the representation of the pressure on the slat it was decided to use the maximum pressure all around instead of the pressure corresponding to the pressure coefficient graph from the aerodynamic study. In that way, by having the highest pressure all around, it is clear that at the end of the structural study the components will resist the highest pressure corresponding to the most critical load case.

There are two distinctive parts to the detailed structural study; it is possible to separate the study of the mechanism components on one side and the study of the LE slat skin and structure on the other sides. That way it is easy to see if the mechanism will resist the different loading conditions, and to see if the slat skin and stiffeners deform adequately.

These two different cases will be analysed and the link between both is done by using some virtual constraints to represent the physical case.

The detailed structural analysis of the mechanism component has not been performed due to the lack of time left at the end of the project. However the methodology has described all the steps to take into account for such a study and this work might be a perfect objective for further work on this concept of LE devices.

Study of the slat skin and stiffeners structure:
Using a CAD design of the skin for the inboard and outboard panels for the LE it is possible to start the structural analysis. From this point it is possible to add the stiffeners (as beams) and to represent each components using different physical representation (Figure 84). For a quicker analysis it has been decided to use a "beam" type of element for the stiffeners and a "shell" type for the skin. That way it is easy to modify the characteristic of each part. The beams can be rapidly changed from a "U" profile to a "T" profile represent a closing rib or other profile and the thickness of the skin shell can be modified. The material properties for each component can also be changed or modified at anytime using the physical properties parameters.

For this project, the material selected for the slat skin is aluminium alloy 2024-T351 with a thickness of 1mm minimum, to 3 mm maximum. The analysis will help to define the optimal thickness, depending on which type of alloy is selected. A starting thickness of 2.03 mm was used as for the research carried out by Ammoo[2].Also the same type of material was used for this simulation in order to represent a material which has good damage tolerance properties and static properties.

Using the same CAD model it is possible to add constraints to represent as accurately as possible how the components will be linked and constrained to the wing box, through the front spar and the deployment mechanism. These constraints also represent how the components are linked to each other and at which position.

For the case study it was decided to represent the mechanism location as a fixed feature for the skin structure, this is done in order to analyse the deflection of the slat. The position of the mechanism and the ribs are taken as fixed edge, this is to show the deformation of the free edges of the sLE device.

Once the constraints and elements behaviour have been described it is possible to introduce the loads applied to the assembly. This set of load can be applied to the different components on different shapes but, to represent the real condition, it has been chosen to apply a pressure on the LE slat skin to see how it will deform.

For this study it was decided to analyse half a panel as the pressure will be applied equally around the panel. This also applies because the panel will have a symmetrical support structure spanwise so the supporting structure will take an equal share of the load. By having a model which is only half of the real geometry this save time during the

calculation process for the FEA software, this also reduce the amount of time for the meshing, or at least to get a quality mesh.

The LE skin was assumed to be fixed through the rib "beam" representation along the span of the panel LE panel analysed. In fact, these ribs (or beam in the mesh) represent the link to the front spar and wing box through the deployment mechanism.

After the representation of the different parameters (load, constraints, physical properties…) has been added it is possible to generate the mesh for the overall assembly and then run the analysis to see what kind of deformation will occur.

Generally this type of analysis for the LE slat skin and structure is important as it allows visualisation of what kind of deformation occurs in terms of bending. The more bending or deformation there will be, the more it would affect the aerodynamic performance.

After all of this it is possible to collect the results and see what the exact deformation is for the slat skin and associated structure.

From then, it is possible to investigate the possible changes for the materials or the thickness and dimensions of the different components. The constraints and sets of load will not be modified, as they represent the real case. Any changes on the materials and dimension will obviously have an effect on the final deflection but also on the overall weight of the assembly. So it is necessary to find a "target" maximum deflection allowable in order to decrease the mass of each component by finding the lowest dimensions which will comply with the maximum deflection.

A critical loading of around 600kg per m^2 which is twice the load Ammoo used for his research[2], was used for this project. It was assumed that the any assembly and panels resisting this level of load would resist any extreme loads during the different flight conditions.

Some of the results for this LE VC panel analysis have been shown as part of the validation process (see Figure 50 and Figure 52 in page 83). As seen on these figures the aluminium skin will deflect by a maximum of 5.36 mm at the middle of the panel section between the fixed edges, these are unsupported edges. This value is quite low considering the use of an extreme load case. Another case was done with a much lower pressure over the panel. A pressure of 100 kg per square meter was used, and the panel would only show a maximum displacement of 1.64 mm. This is extremely good results, as this is in line with the kind of tolerance problems met by the designers when designing this type LE device assemblies. Such a low displacement will not have an impact on the aerodynamic efficiency of the wing.

However due to a lack of time at the end of this research project it was not possible to fully optimise the structure in term of section, thickness and material choice. It has to be noted that the given methodology provides all the required explanation to analyse this structure further.

5.7. Integration of aerodynamics and structures

Both the aerodynamic study and the structure study have provided some results. Using both sets of results it would be possible to implement a totally new deployment mechanism in order to obtain a VC LE device solution. Some of the early results have shown that there is enough space a fit the mechanism, and the CFD simulations have shown that the deployment does have a positive effect on the performance. Even if all the structural analysis has not been fully completed it shows that a possible aluminium skin could in fact sustain the kind of loads applied to the LE during the different phases of flight. Also, both parts of the methodology have proven to work perfectly as a parallel process, with each of them bringing new results to the other part and making this a good combination.

5.8. Detail assessment of mass, reliability, and cost estimation

The detailed assessment of the possible overall mass of the LE device, plus the reliability of the system and finally the cost is possible to be calculated using all the data provided during the different analysis phase. However, due to the lack of results for some part of this case study it is not possible to do the detail assessment for the mass, reliability and cost.

5.9. Case study summary

It has been demonstrated that it is possible to put the new design methodology into practice and obtain interesting results. The case study provided a perfect example to run this design methodology and see how each part of this methodology can work for future cases.

Also the results provided by the aerodynamic study have shown that VC LE devices could improve the performances compared to the neutral airfoil

The mechanism created during this case study has shown to be following the deployment trajectory and had been optimise to achieve an ever better deployment. For this mechanism it is also important to realise that it will not intrude with fuel tank in the wing, which make this design even a better solution compared to slat track mechanisms.

It is also very interesting to see the kind of results generated by the different tools as they provide important results and also shows the different trends for the “running and the making” of VC LE devices, as well as the financial implications.

More detailed comments on the results and the methodology can be found in the next section.

Chapter 6

6. Discussion

6.1. Introduction

The focus of this research was on the development of a design methodology for the design of LE devices applied to VC concepts. This methodology covered the whole design process from initial inputs to flight tests. However for this research the emphasis was on the initial design phase and not the latest parts of the methodology (prototypes and flight test).

A case study has been used for the different areas of investigations and the use of innovative tools for the design of such complex devices.

The different areas of investigations included:

1) Design methodology

2) Mass and reliability estimation

3) Aerodynamic analysis of the airfoil (different speed and flight conditions).

4) Preliminary mechanism design for the deployment mechanism

5) Initial structural analysis (under different loading conditions)

This chapter "groups" all the different aspects covered during this research project. It also describes both, the problems encountered by the author for this study, and the possible superiority of using such an innovative concept (i.e. VC LE devices). For each parts mentioned in the above list, there will be an extensive description of the advances made during this research project as well as comments on the limitations and advantages of the provided methodology and case study results. Also there will be comments on the the result of the case study compared to the results of the estimation tools developed for this research.

This discussion will be used to explain in detail the advances made during this research and also new and innovative design solution and tools created.

6.2. Overall design methodology

The design methodology developed in this thesis is quite innovative on the way it is structured as well has being driven by specific objectives (mass, reliability & cost) from an early point. Another important feature of this methodology is the use of optimisation tools and state of the art engineering softwares. Excel based tools have been developed by the author have also been used extensively to further improve the design process. It is important to notice that a normal design methodology would take into account the overall design of the aircraft from the conceptual level, but this methodology would focused only on the design of LE device but shows the effect of it on the overall aircraft. This limits the scope of the design area and also adds specific constraints to the design methodology to drive the user to generate an optimum result, for LE devices.

A limitation of this methodology is that the user needs to be experienced with the use of the different tools for the design/analysis/optimisation. This experience, or learning curve to become experienced, can be a difficult process. This could be a disadvantage as the user might lose essential time when learning the overall process and might not design any new and innovative components. This also means that users have to fully understand general aircraft design and the implications of the results created. It is important to remember that the current industrial approach to complex problems is to have different teams looking at the aerodynamic, the structure, the kinematics…all separately which is the contrary of the proposed methodology in this report.

Another limitation of this methodology is the extensive use of engineering software and simulation. This can be overexploited and use too much computational time. Taking the case of the CFD calculation for example, one case can take up to two or three days for a viscous case at high speed, and this was only for 2D cases. A possible 3D case could take more than twice as long. This would obviously increase the time to design, or require investment in better computational capability.

It has to be kept in mind that the overall objective of this design methodology is to produce LE devices using a faster process, which provides reliable and practical solutions. Overall, engineers using this methodology will always have to remember that the main objective is to design the lightest possible solution which still complies with the loading cases representing the different flight conditions, and the different regulations.

Looking at the results obtained by doing the case study it is clear that this methodology can work, and can produce coherent and good results. Each step of the methodology provides results which help to progress towards a reliable, light, cost effective and easy to maintain design solution.

However, there are a few points which could be developed or improved on future research based on the limitations of this methodology.

One part of this design methodology could have been further improved, but due to the time and financial constraints it was never possible to think of doing a shape optimisation

for better aerodynamic performances. A full shape optimisation program in relation with the given methodology could radically change the overall results as it will drive directly both the aerodynamic airfoil profile and the deployment mechanism at the same time. This could also include full wind tunnel testing. The shape optimisation of the airfoil using advanced CFD and experimental techniques might increase the advantage of using VC devices at the LE if it was linked to this design methodology.

This design methodology did not also consider a full study of the electronic and systems behind the deployment of the LE device. A study of the different types of systems and actuators could further improve the efficiency and mass of the overall LE mechanism. This part was out of the scope of this research project, due to the time limit. However, for the different estimation tools developed during this research project, assumptions were made in relation to existing designed and already used actuators to represent as realistically as possible the real components. Limitations on this will be discussed in the following pages.

Another part of the process which was not achieved was to design a full wing including a prototype of each of the sections to see the full deployment mechanism. That would have helped to observe the real behaviour of the different mechanisms. In order to achieve such a large testing procedure it would have been necessary to secure backing from an aircraft manufacturer willing to invest money in this type of research.

However, for all the limitations or disadvantages this methodology could prove to be a really useful tool if used properly. Indeed, it would be possible to develop new mechanism, study VC concepts, or even generate fully optimised designs. Also it has to be noted that part of this methodology can be used on their own to generate specific and isolated results.

This methodology is a totally new tool which covers the overall design at the conceptual stage. Researchers in the past have analysed different LE devices, but none looked at developing a specific design methodology for the design of these devices from the conceptual stage, and the implications to the final result. Most methodologies have been developped for the design of general high lift devices (LE and TE) as explained in chapter 2, but none provide such a new way to design LE devices. This research has proven that a new way of considering the design of LE devices, including VC options, can be used to lead to better design solutions along with optimised deployment mechanism and better understanding of the cost, reliability and mass of the LE devices.

This methodology could be seen as a breakthrough in the way aeronautical design offices “tackle” thez design of LE devices. Also, to add to this, this is one of the first research projects to consider the design and effect of VC option for LE devices.

6.3. Discussion on mass, reliability and cost estimation

The following paragraphs will present the advantages and drawbacks of the mass and reliability and cost estimation tool. There will also be comments on the problems which occurred while using these tools as well as comments related to the efficiency of such tools.

6.3.1 Mass estimation

The mass estimation tools produced some really accurate results, and also provided estimation for a wide range of different mechanism types. The results obtained for the case study can be taken as a pretty accurate estimation of what the real mechanism would weigh. A new VC concept (following this research) should prove to be just as light as other concepts and still produce the required VC solution. The estimation tool allowed doing a rapid estimation of the mass for different types of configurations; this gives a very quick overview of all the different possible solutions, including the different number of panels within each wing section.
Within the given methodology, it is important to understand that the mass will be reduced using optimisation during the mechanism design and structural analysis phase. The results obtained for the mass estimation are supposed to be a "rough" idea of the final value, however using the results obtained in the validation of the mass estimation tool it is expected to be within 10% of the real result.
The final result for the overall mass compared to the mass estimation at the early design stage will be discussed in the next paragraphs.

The mass estimation has given rapid and clear results. However this tool has one drawback, at the present time it cannot change simultaneously the number of supports per mechanism and the mass ratio of each mechanism. In other words, when increasing the number of mechanisms, for each panel, the mass ratio of each mechanism does not decrease. That means that this tool "carries" a higher mass when increasing the number of mechanisms. This could be further improved in future work. But for this research this offset error was similar across the different type of devices, so the error would be equal. This represents an offset of the result more than an error due to the equivalent offset value.

It is also important to know that the lack of available data related to the mass of the different components used in modern aircraft (A380, B777) has an implication on the quality of this estimation tool. The author only had access to data related to "older" aircraft and so these aircraft do not use the latest technology in term of materials and structure optimisation. This lack of available data is due to the fact that the data on modern aircraft weight are still confidential and could not be accessed by the author at the time of the research.

Despite the given drawbacks due to the initial lack of data, this mass estimation is an innovative tool which provides accurate mass average results. Also, one of the major advantages of this tool is the simplicity and flexibility to generate customised results

within a very short amount of time. In the past research focused on the mass estimation for the wing box but none developed a specific tool for LE devices mass estimation. Moreover, it has been impossible to find existing tools or formulas giving mass estimation for VC options.

6.3.2 Reliability estimation

The reliability analysis has given some results, as quick and clear as the one for the mass estimation. This tool has proven to be really useful for an early estimation of the overall reliability for different types of devices. The advantage of this tool is its simplicity and flexibility; it quickly gives a set of results to compare different types of devices. Also this tool has been validated against known reliability calculation methods and systems and has given coherent results, which should give confidence to future users of this program.

The use of the reliability tool has shown to provide a good set of results for the case study. It clearly demonstrated that using different configurations of devices at the LE can increase or decrease the failure rate by up to five times compared to the least reliable configuration. This shows the importance of reliability when considering the design of LE devices at the concept stage.

One drawback of this estimation tool is the lack of data regarding the different components' reliability. There was not much data accessible by the author regarding the reliability of components used on the LE devices. The main data available, on which this tool is based, was related to generic types of components (under specific conditions) from the Non-Electronic Part Reliability Data [47]. The same type of tool already exists but not specifically for this type of application. This existing tool (or software) provides a detail reliability analysis for complete systems, but it does require a long time to define the system analysed. So it is much easier to use this new estimation tool for the design of LE devices instead of using other generic reliability softwares which normally requires a high level of detail reliability knowledge and generally only represents detailed reliability for different components.

Another drawback of the reliability tool comes from the fact that all the electronics and systems study has been omitted from this research. The reliability study might have a better and more accurate use if it included data on the actuator and electric systems. The focus was on the general mechanical design for this type of concept, and not looking at the actuation systems in more details. The actuation systems and the electronics to command then are usually considered when studying the reliability. In current designs it is quite common that most of the main parts of the mechanism are designed for not failing at all during the life of the aircraft. Only smaller parts are considered to be changed due to wear and tear. Also, most of the modern LE deployment mechanisms use a fail safe solutions with different degrees of safety. This would obviously affect the reliability if incorporated in this tool.

Overall, the reliability estimation tool has proven to be really useful. It provides simple results which allow to make rapid judgment on the reliability for different configurations. This tool is also totally new, as it is the only one which provides tailored results for LE

devices. It means that someone not being an expert in reliability would be able to generate useful results. It will enable users to quickly analyse the results. This makes this tool a very important addition to the study of concepts for LE devices design.

6.3.3 Cost estimation comments

The cost estimation tool has shown to give good results for the different parts, as it covered quite a wide range of cost issues.

It is very difficult to analyse the exact validity of this new tool for two important reasons. It is difficult to represent the cost of different types of assemblies as well as using parameters to describe the dimensions of these assemblies (i.e. the different LE concepts). Another problem is the actual cost data for the manufacturing and the raw material, this is associated with the "buy to fly" ratio, which none of them are publicly published and/or very well understood by the aerospace industry.

However, the author has tried to make this problem have a low impact on this research. Indeed, the author has developed a very flexible tool which is already made to welcome change for the values of the different inputs. This is especially important as the cost of material to purchase is changing at a relatively high pace. This is depending on worldwide economy and the balance of supply and demand. This is extremely important to consider as the aerospace industry as a whole is demanding more and more composite material and titanium, which then has a detrimental effect on the purchasing price per kilogram.
The other interesting point to note is that the author, through the structure of his tool guarantees that if there is an error on the estimation of cost for one material or one process, this error is spread across the tool and the so the results generated will only be an offset of a more accurate value. This makes the error equal for the different types of devices and for the different sections (inboard, central and outboard).

The manufacturing cost estimation tool has shown that it is possible to estimate the cost of producing the different type of LE devices and gives results showing an expected trend but giving the users more detail on the manufacturing cost. This is an innovative tool and no other researches have developed such a flexible and user friendly tool, specifically applied to LE devices design.

The part of the tool focusing on the cost of fleet management is also really important as using different devices might have, in fact, a totally different impact (financial) on the airlines as they use different flight patterns. The case study has shown that it is possible to use this tool for a real practical case and for the users to see which device to select or what flying pattern to choose for their specific device.

6.4. Discussion of the aerodynamic study

To show the effects of a possible variable camber it was necessary to run an aerodynamic simulation. Initial cases were run at different speed to represent the different flight conditions. The results have shown that a VC LE device could be beneficial to the overall aerodynamic cruise performance, through generating extra camber and more lift than a "clean" airfoil profile.

The results obtained are assumed to be realistic due to the mesh quality and the validation process used.

The study for the low speed cases has proven that the overall performance of the airfoil used in combination with a possible VC solution of the LE is improved compared to the original airfoil. The results have shown in improvements of the L/D ratio for the landing and take off case.
The low speed cases have been carried out using a software which has somewhat limited capabilities, for example Xfoil could not represent an airfoil with a separated components including a gap between these two components (typical slat concept). This limitation implied that it was not possible to carry out a study to compare the newly design VC concept with the more classical slat track concept.
On the other hand, the Xfoil program has been used extensively for this research and has shown to give interesting results for specific case. Also, an important point is that this program does not require too much time for computation of the results and the input data can also be generated within a short amount of time. Relating to this, the downside of using Xfoil is the somewhat reduced accuracy of the results and the lack of control on the representation of some physical characteristics of the air flow. Also the poor accuracy of the mesh can to some extent be an issue to the user as the software uses a panel technique to mesh the airfoil. Another downside of Xfoil is that it does not represent the behaviour of the surrounding air.
The user must be careful on how to use this software and try to use it for specific test conditions linked to the simple capabilities of the software.

The results provided for the high speed case are interesting as they show that it is possible to get an improvement of the wing efficiency by using a VC LE device. However, as explained in the case study, the deployment of the VC LE device will have to be considered with attention as it could be having a negative effect.
As explained before, the increase of angle of attack will ultimately generate more drag, even with a 2 or 3 Degrees of camber at the LE it is possible to improve compared to the original airfoil, but these improvements are degrading as the camber and angle of attack increases. This monitoring of the deployment could be controlled through the flight control software link to the LE device deployment actuators. Also the flight control computer will have to directly related the angle of attack to the camber angle at the LE in order to provide the pilot with a beneficial configuration.

Another point to consider is the time required to run any simulation, which for this research did not consider a 3D case. This is an important factor to consider when doing this type of research.

Due to the aim and scope of this project, the study of the combination of a LE and TE VC wing was not studied. Although the results obtained in this research are interesting, they might be of a greater importance if it was shown that the use of a TE VC in combination with a LE VC device improves the performance of the wing section. In fact, based on the research carried out for the LE it might be able to apply the same type of methodology for the design of TE devices.

It has to be noted that the results of the different analysis have shown an improvement on the original airfoil profile used for the case study. it has also appeared that the VC LE does improve the performance in a similar fashion as a more classical slat concepts, However, because of the slot in the usual slat track concept, the stall is delayed to a higher angle of attack which is not always true for the VC LE device (see figures in 176). So the landing and take off performances might not be as good as the more classical slat track concept, but it provides on the other sides benefits for in cruise condition improvements. This might be financially more profitable for the airlines.

However, it is true that experimental results would be necessary to further reinforce the confidence on the simulation results. Experimental results could not be generated through wind tunnel testing (or flight test) due to the financial cost of hiring the facilities and producing the models. Also during this research it was not possible to study the effect of the structure deformation during cruise condition. This would change the LE shape and to some extent would modify the aerodynamic performance.

6.5. Discussion on the mechanical and structural design

6.5.1 Mechanism design

The mechanism study, which included the use of new software, allows users to develop innovative types of mechanisms in a shorter amount of time. However there are some constraints linked to the SYNAMEC software (and its specific version) used to design mechanisms. At the time of the research it was not possible to input restrictions on the available space at the synthesis stage. In fact, the generated mechanism could sometime move outside the "working envelope" and could clash with some other elements. That also means that the user of the software has to do some more work when getting a solution which clash with other elements. One of the options is to get a new mechanism and then put a distance constraint compared to another element to avoid any contact between these elements. That also means an increase in the time to design and add extra variables for the optimisation.

Another drawback is the amount of time and effort it takes the user to become familiar with using the software in order to get the best design quicker. This means the user has to get a "feeling" or expectation of what type of solution will be generated and what are the problems likely to happen. With experience the user will be able to use different parameters, or a larger range of variation for these variables. This will then create an improved design solution which is suited to each individual case.

Even if there are drawbacks, it is still very interesting to use the Synamec software to do the mechanism synthesis as it can help to generate quick solutions to fit the required objective. This tool, if used correctly, can help to get a quicker solution. An experienced user could generate a few solutions also investigating different positions for the main fixing points within the design space.

The use of the optimisation tool also increases the time to design the final virtual prototype, but overall it should reduce the overall time to get the perfect design. In fact, by running the optimisation it would be possible to reduce the overall weight and still get a structurally safe design solution.

For the case study the SYNAMEC software has shown to give good results. Using this tool it was possible to generate deployment mechanisms (or kinematic chains) which adapted and related to the input specified by the user. This gave flexibility to tune the mechanism to get a better and more accurate solution. This could be further improved by using the optimisation tool, which one could automatically optimise the solution. Results from the case study have shown that this optimisation phase is important to get the "best fit" type of results. This optimisation takes into account the user-defined variation range for the different parameters (fixing points and deployment points positions).

The given time for this research project was not enough to consider a full 3D case as more problems would probably arise from this. It is possible to consider this as a full research project in the future.

6.5.2 Structural study

The structural analysis has given some results regarding the integrity of the overall system. This has been proven by keeping an organised and structured mesh on the different components and by representing the different types of contacts. The results obtained show that it is possible to design a set of LE devices using a VC concept. Using optimisation and trying to find the optimum solutions for the values of the different parameters it was possible to produce a LE device which will not deform too much under the different loading cases.

For the structural analysis it was not possible at the time to do the analysis, but it is expected that the different components of the mechanism will resist the load as if required the material to manufacture these links can change. However, it has to be noticed that the cost implication on changing material might be quite high.

For the study of the slat element, and especially looking at the structural integrity of the slat skin and ribs, it was shown that it is possible to make this type of device. The arrangement of the stiffening structure was also studied and shown that the overall structure will be strong enough to resist the applied loads. The study has shown that the maximum deflection was within some acceptable range (less than 10 mm or 2% of airfoil depth), and that deflection should not have a major influence on the aerodynamic efficiency of the wing.

Due to the lack of funding and time, it was not possible to design a full structural test to see the deflection of the slat under some experimental loading and compare this to the simulation results. However the analysis was done using different types of settings for the constraints. These different constraints were representing the different possible settings for the slat element as well as the structural elements (stiffeners). It also has to be noticed that in the validation process, it was shown that the analysis was done with a good understanding of the potential problems as well as an accurate description of the possible constraints. The validation chapter has shown that the author was able to control the meshing process, and the mesh quality. The author was able demonstrate at what points the mesh was accurate enough to provide the best description of the physical deformation.

Also it might be necessary in the future to investigate the structural integrity of the joint between the lower wing cover and the bottom edge of the LE device.

6.6. Comments on the discussion

It is possible to give a short summary on the different points mentioned in this discussion section related to both the methods and the results created during this research.

As any research associated with design of new device for aircraft, it has proven really difficult to access data (mass, reliability, materials, etc…) related to contemporary aircraft. Despite many efforts from the author to try to gather this type of data, it was very difficult to obtain them. This is a major drawback as it does influence the overall methodology.
However, the author used a comprehensive set of validation steps in order to prove the quality of the obtained results. This is why this research could be developed further if used by a major aircraft manufacturer having existing data. Also it is safe to assume that the different estimation tools can be used with a good level of confidence on the generated results.
Also it has been shown with the different estimation that the VC LE devices does indeed compare quite well compared to the more classical concepts. As seen in the different graphs the cost or mass of the slat track can be to some extend more expensive and somewhat heavier than a possible VC LE device.

Due to the lack of practical experiment it is quite difficult to see the real problems which could occur with the given new design. This means that the lack of experimental results is detrimental to the quality of the results obtained in this research.
This especially applies to the problems which would be expected to appear when doing the final assembly of the newly designed LE device. Problems might also occur during the deployment phase. Since the research was mainly done using simulation tools it is very difficult to predict the problems likely to occur in reality.

On the other hand, the validation process for this design methodology has been as accurate as possible, to show that each step of the design and simulation process is well controlled. These validation processes, in relation to existing results and best practice, should give a high level of confidence in the result obtained by the different tools used in this methodology.

Another very important point which appeared during this research is the amount of time required for any user to run efficiently all the different tools used in this methodology. If this methodology was to be used in a large aircraft design office it would be expected that different people (experts in their own field) would use different parts of the methodology and so each of them would be experienced in using different software. Also it would be necessary to change the usual organisation of the design office as this methodology has direct implications to the different sections of a typical design office (weight, stress, cost, aerodynamics…).
This might require a change of culture and organisation within an aircraft design office, which might be difficult to achieve. Indeed, current aircraft manufacturers tend to organise their design offices in separate “skills team” working on a same aircraft programme. This is why this methodology could be quite difficult and complex to

implement in a large aircraft design office due to the radical change in the way the design of LE device is done.

Following on these comments, and the previous paragraphs discussing the effectivity and efficiency of the different tools, they show clearly that this research project provided some important results. Both the innovative design process and some of the findings add to the research and recent development on aircraft design and make especially to LE devices design.
The methodology has also proven to be useful for generic LE devices design but also for VC applications, considering this under different aspects. This type of study has never been done before, and is innovative and efficient in many ways.

Chapter 7

7. Conclusions and recommendations for future work

7.1. Conclusions

1) The aim for this research was to develop a full, and innovative, design methodology for the design of LE device, specifically looking at the design of LE devices using a VC concept. The methodology covered both the aerodynamic and Mechanical/Structural implications. .The methodology is driven from an early stage by the different estimation tools to lead to an optimised solution, based on initial requirements, including cost, maintenance, or overall mass. The methodology has been fully described and a case study (the ATRA aircraft) has been used for the examination of this design methodology. This case study has shown to give acceptable results and shows that it is actually possible to design a new LE device concept (using VC) and shows the benefits. Overall this methodology if used properly and using all the tools mentioned in this report could aim at the production of innovative solutions. It should also save time during the design process, and finally it should provide a more viable design at the end of the process.

2) The mass, reliability, and cost estimation tools have proven to be a basic tool which give a good idea of the expected value for the different results. The results given have been compared with existing aircraft and show that the tool gives coherent and reliable results. Users can be confident that this tool will give them a set of valid results in a small amount of time. These different tools can also be used on their own for specific studies.

3) The aerodynamic study has shown that the new LE device using a VC concept does improve significantly the L/D ratio, especially for the low speed case (for landing and take-off). However, any use of VC LE devices also have to be considered carefully as it can be detrimental if use at a angle of attack too high. The High speed case study has also shown similar improvement compare to the original airfoil, but it does also require careful monitoring for the angle of attack.

4) The proposed innovative mechanism design to deploy the LE has shown to be able to meet nearly any type of “near” circular trajectory. It is interesting to see that such a design could actually provide an answer for the deployment of most of the LE systems actually used on commercial and fighter aircraft. Another point to notice is the use of totally new technology for the design of such a complex deployment system.
Using the case study it is clear that using engineering software reduces the time to design, and the mechanism to satisfy the deployment trajectory has been improved compared to the earlier trial and error (more classical) method. Also the proposed mechanism do not intrude the fuel tank in the wing box which is an important point as there will be no need to weakened the front spar.

5) It has been shown that a given configuration for the slat skin and structure can resist to the different loading configurations (depending on the different flight phases) based on a critical loading condition. The case study and the associated FEA results have shown that the structure of the overall system will keep its structural integrity. Also all the methods for the analysis have been validated using the different validation methods.

7.2. Recommendations for further work

1) The methodology described in this thesis is a foundation work for further development for the design of LE devices, and especially LE devices including a VC concept. The methodology has been developed with a regional transport aircraft in mind and so further details could be added to apply this methodology to other aircraft concepts (such as fighter aircraft, ultra large commercial aircraft, freighter…), which would require the use of moveable LE devices.

2) The mass and reliability assessment tool could be improved on two ways. It could be improved in taking into consideration the trailing edge concepts (and maybe ailerons) to give a better idea of the whole wing mass at an early stage of the design process.
The tool could also be improved further by having a lot more available data related to the mass of the different elements or the reliability of these elements under different flying time and conditions.
Another area which could be investigated would be an associated tool on the cost of manufacturing for the different components. That would give a better idea of the cost of the different concept and the manufacturing process involved with them.
It could be also interesting to develop an optimisation tool for these different modules to find optimum configurations depending on the input data.

3) In terms of the aerodynamic study, it would be necessary to do further work on the effectiveness of the LE device and especially looking at a full 3D CFD analysis at the different flight conditions. However such a study might take a long time (at the present day) compared to the amount of computation required to solve a full 3D flow analysis for a viscous case. It would also be necessary to take time to study and analyse the real effectiveness of this LE device concepts using wind tunnel testing. Sadly due to the timescale of the project as well the financial limitations it was never possible to do such a testing phase. Also it would be recommended to investigate further and maybe optimise the position of the LE rotation as this was not possible in the timescale of this research.
Finally, another area of to investigate would be the optimisation of the airfoil surface and also the study of the combined effect of VC LE devices and TE VC device together. More investigation on the possible development of a fully cambered wing might prove to be of an even greater benefit.

4) In terms of the mechanical study, it would be necessary to do a full 3D scale model of the deployment mechanism to refine it and also to see if there are any problems during the deployment. This might especially happen under some of the high loading conditions.
Another aspect of the mechanical and structural design could be further improved by doing a full load test on the slat skin when using flexible skin (if used) and do more research on alternatives materials. As other materials would have implications on the skin roughness and so the aerodynamic efficiency of the wing.
Other areas of research could focus on the study of the material and structure used for variable camber including study for bird strike and lightning resistance as well as a full vibration and noise analysis. This, in fact, might be detrimental to the environment and might not comply with current regulations, hence why it would be important to analyse it.

And finally for it could be interesting to look at the configuration, design and structural analysis of the joint between the bottom wing cover and the bottom edge of the LE device, as possibly more design solutions could be developed.

5) One area which has not been investigated in much detail in this report is all the systems and electronics including the slat drive systems. Much emphasis will need to be put onto this aspect, as it could be a critical part of the design. Further work could also be done on making the flight control computer to automatically change the camber through fly-by-wire systems of the wing whilst flying.

8. Appendixes

8.1. Reference List

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8.2. Appendixes A

Appendixes A

		ATRA-80	ATRA-100	ATRA 130
Number of Passenger		83	108	133
Range [nm/km]		200/3706	2250/4170	2500/4633
SPEED, MACH				
Landing (max.)	[M]	0.2	0.2	0.2
Take-off (max.)	[M]	0.2	0.2	0.2
Cruise	[M]	0.8	0.8	0.8
LIFT COEFFICIENT, C_L				
Landing (max.)		2.9	2.9	2.9
Take-off (max.)		2.2	2.2	2.2
Cruise		0.49	0.51	0.53
WING (A320 Type)				
AR		9.5	9.5	9.5
Taper ratio		0.274	0.274	0.274
0.25 Sweepback		25	25	25
Span, m	[m]	32.357	32.357	32.357
Area, S	[m^2]	110.21	110.21	110.21
Mass, m	[kg]	5811.13	5811.13	5811.13
Root chord, C_r	[m]	6.673	6.673	6.673
Tip chord, C_t	[m]	1.586	1.586	1.586
Mean aerod chord MAC	[m]	3.439	3.439	3.439
t/c root		13%	13%	13%
t/c tip		9%	9%	9%
trailing edge flap		Single slotted	Single slotted	Single slotted
leading edge flap				

Table 7 - ATRA Specifications[2]

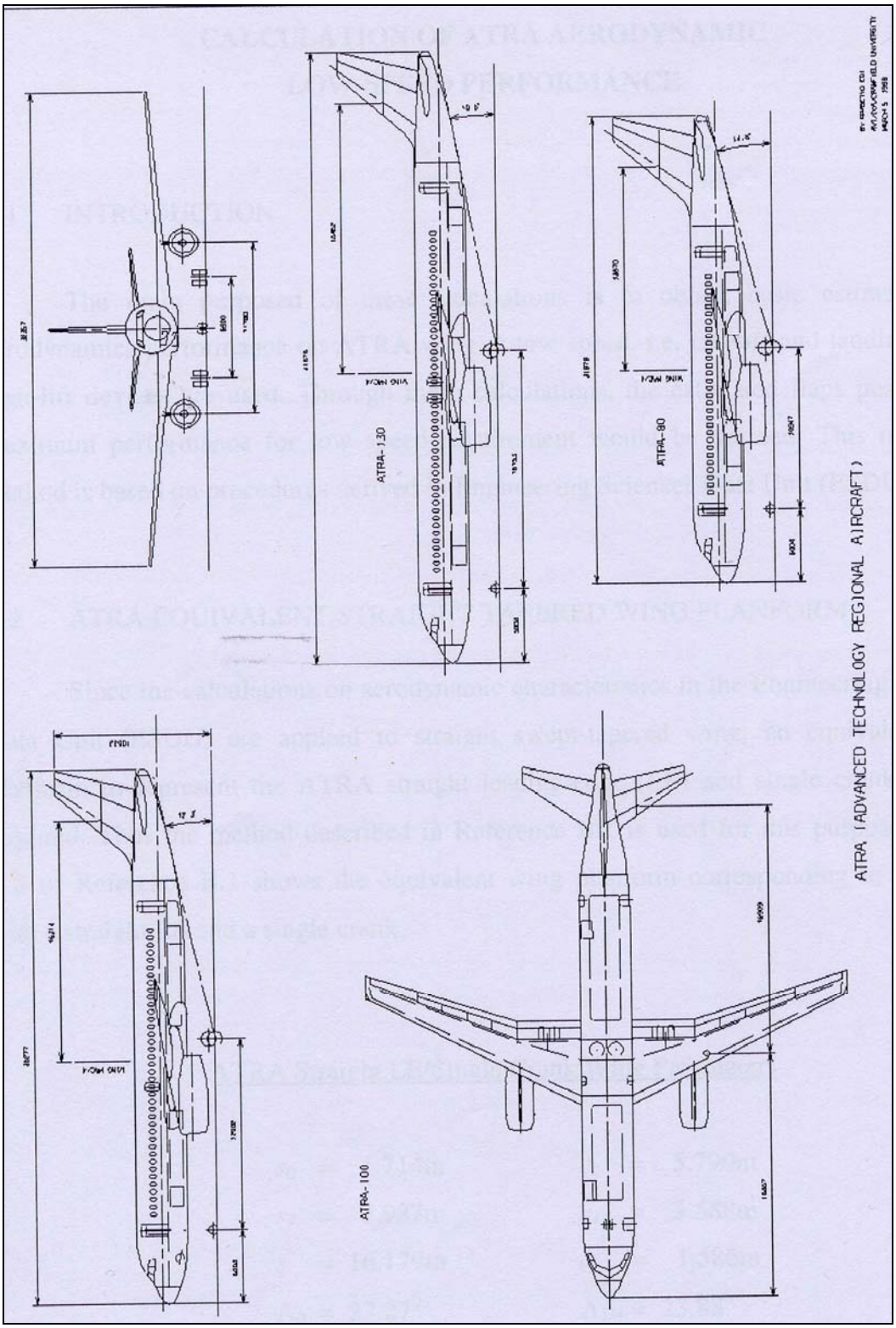


Figure 85 - ATRA family [2]

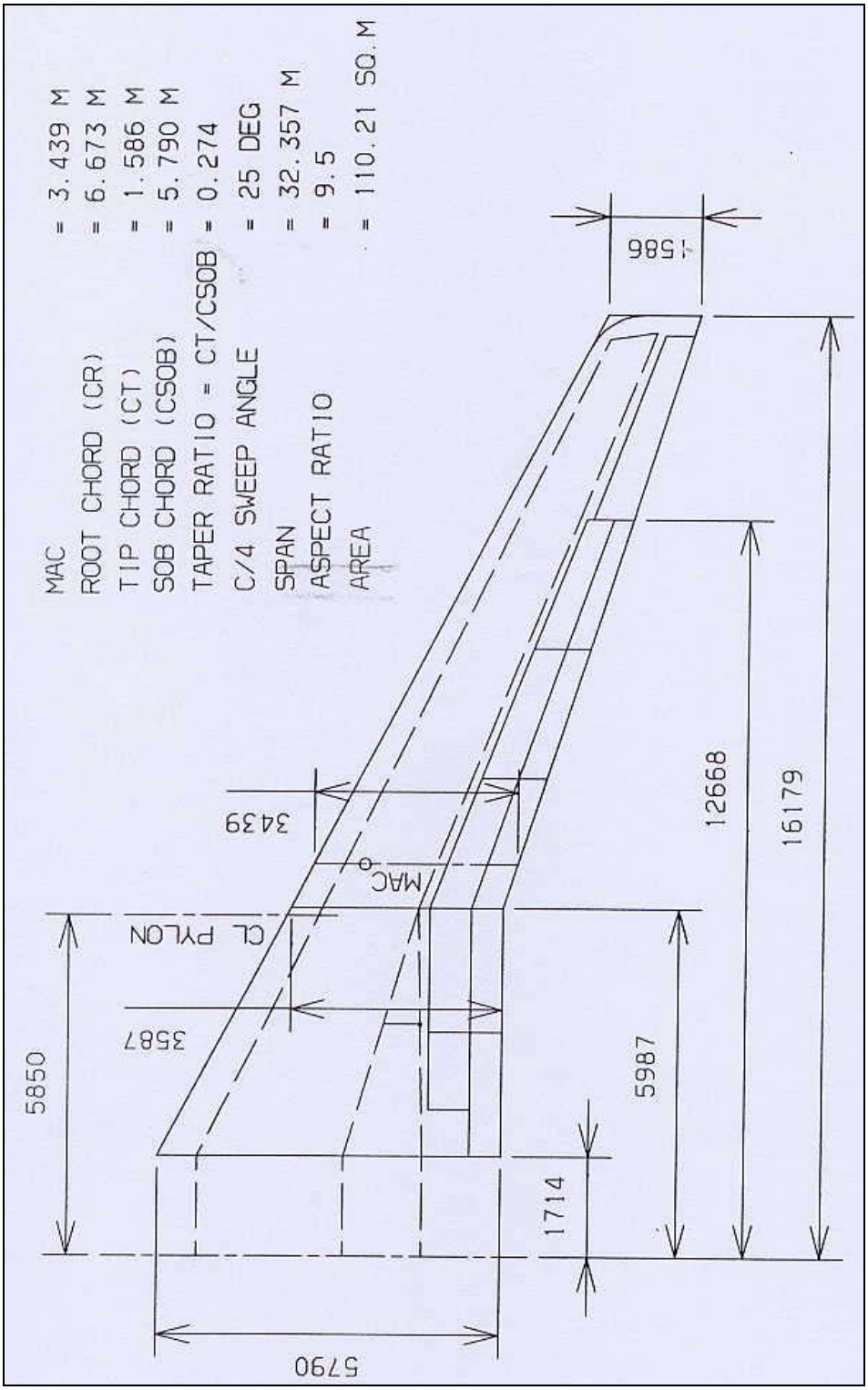


Figure 86 - ATRA wing geometry [2]

8.3. Appendixes B

XFOIL 2D Descripition
XFOIL is an interactive program for the design and analysis of subsonic isolated airfoils. It consists of a collection of menu-driven routines which perform various useful functions such as:
Viscous (or inviscid) analysis of an existing airfoil, allowing
forced or free transition
transitional separation bubbles
limited trailing edge separation
lift and drag predictions just beyond CLmax
Karman-Tsien compressibility correction
fixed or varying Reynolds and/or Mach numbers
Airfoil design and redesign by interactive modification of surface speed distributions, in two methods:
Full-Inverse method, based on a complex-mapping formulation
Mixed-Inverse method, an extension of XFOIL's basic panel method
Airfoil redesign by interactive modification of geometric parameters such as
max thickness and camber, highpoint position
LE radius, TE thickness
camber line via geometry specification
camber line via loading change specification
flap deflection
explicit contour geometry (via screen cursor)
Blending of airfoils
Writing and reading of airfoil coordinates and polar save files
Plotting of geometry, pressure distributions, and multiple polars

Simple models
Quick results

Generate 1st profile
Get rough estimation of Cl Cd drag polars Cp…
Quick result easy to analyse

8.4. Appendixes C

Analytical description of the mass estimation tool

This tool is structured on the two main layers (described above), the aircraft layer and the type layer.
However, the structure of the calculation has many more layers, each of them is described below with the related equation as well the related parameters to resolve each equations.
The following is the analytical description of the calculations taking place step by step in Figure 87 (in page 165) and Table 1 (in page 48), and how the results are generated.

At "aircraft level" the results generated is the LE system overall mass for the whole aircraft (as being two wings):

$$M_{LE,total} = 2 \times M_{LE,W}$$

Equation 8-1

With:

$M_{LE\ total}$=	Mass of the LE system for the aircraft	[kg]
$M_{LE\ W}$ =	Mass of the LE system for one wing	[kg]

This is the "wing level" equation. The mass of the LE system for one side of the wing is calculated as follows:

$$M_{LE,W} = M_{LE,Inboard} + M_{LE,Central} + M_{LE,Outboard}$$

Equation 8-2

With:

$M_{LE\ inboard}$=	Mass of the LE system for the inboard section	[kg]
$M_{LE\ Central}$=	Mass of the LE system for the central section	[kg]
$M_{LE\ outboard}$=	Mass of the LE system for the outboard section	[kg]

If the aircraft only has two sections (inboard and outboard) the user only has to specify that there is zero panel for the central section, in that way the mass of the central section is equal to zero.

This is the "wing section level" equation. Each of these sections is calculated using the same kind of formula:

$$M_{LE,Section} = \sum_{i}^{n} M_{Di}$$

Equation 8-3

With:

$M_{LE\ Section}$=	Mass of the LE Devices of one section (in/outboard)	[kg]
i=	Device types	
M_{Di}=	Mass of each types of device	[kg]

$M_{LE\ Section}$ can be either $M_{LE\ inboard}$ or $M_{LE\ Central}$ or $M_{LE\ outboard}$ depending on which section is studied.
Each of the devices mass is calculated using the number of panels using each type of device. This is the "type level" equation:

$$M_{D,d} = N_{P,d} \times M_{P,d}$$

Equation 8-4

With:

M_{Dd}=	Mass of one device	[kg]
N_{Pd}=	Number of panel of each type of device	[kg]
M_{Pd}=	Specific panel mass for each type of device	[kg]
d=	type of device designator (Krueger flap, Slat track, VC …)	

The "panel level" equation defines the mass of each panel, as being the sum of the mass of the different type of components:

$$M_P = \sum_{i}^{n} M_{COMP,i}$$

Equation 8-5

With:

M_P=	Mass of one panel	[kg]
M_{COMP}=	Mass of one type of component per panel	[kg]
i=	type of components designator (bearing, actuator, flap…)	

The final equation is the one defining the "component level". This equation defines the mass of each panel depending on different mass parameters (described below):

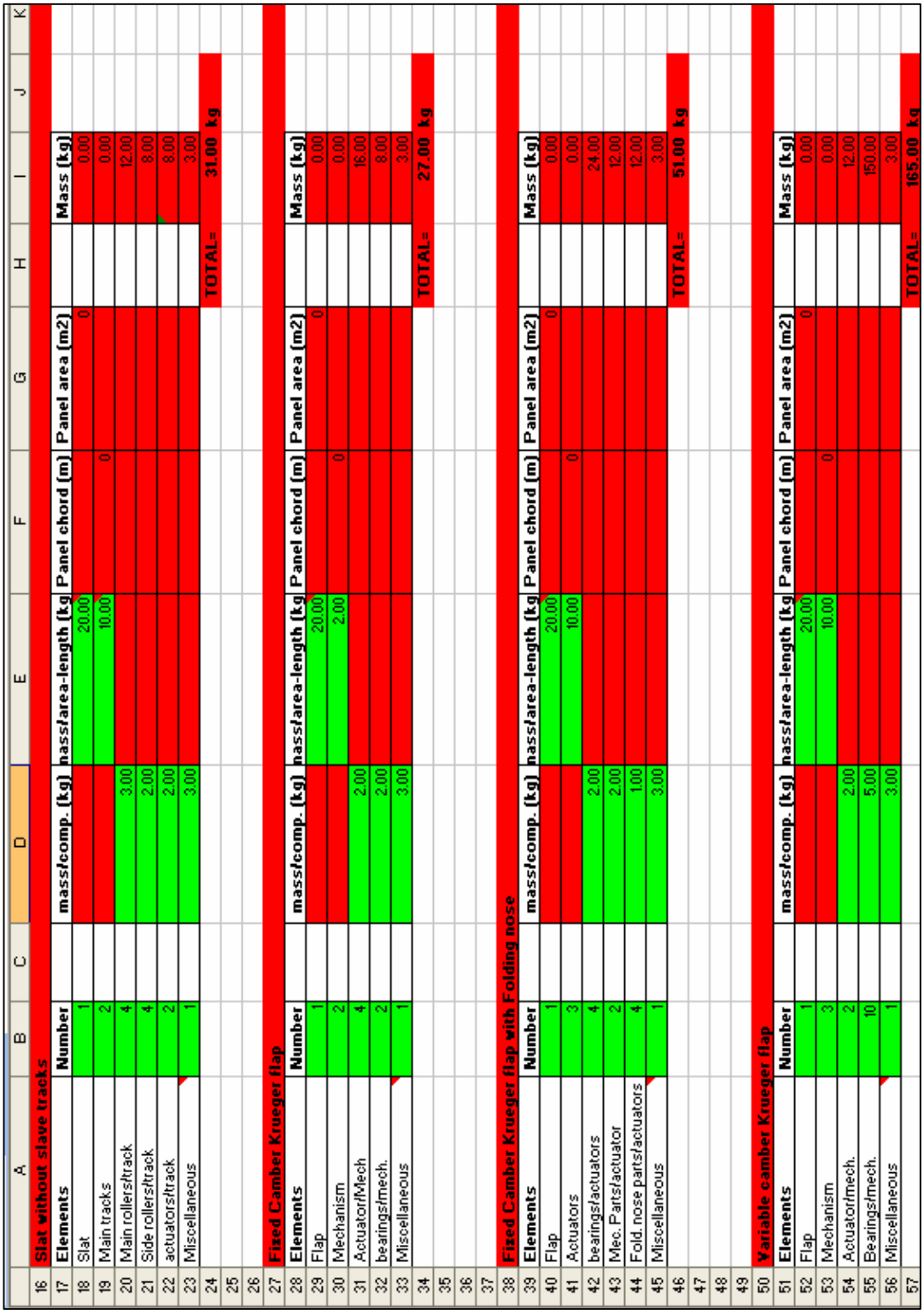
$$M_{COMP} = \alpha(M_{COMP,Specific} + M_A.PA + M_c.c_{average})$$

Equation 8-6

With:

α=	Quantity of component	[--]
$M_{COMP,Specific}$=	Specific mass of component	[kg]
M_A=	Specific mass ratio per unit area	[kg/m^2]
PA=	Projected panel area	[m$^{2]}$
M_C=	Specific mass ratio per chord unit	[kg/m]
$c_{average}$=	Average panel chord	[m]

It has to be noticed that for this "component level" equation, only one of the three categories can define a component type mass. For example, actuators have a specific mass component ($M_{COMP,Specific}$), when a flap mass will be define by the product of the flap specific mass ratio per unit area (M_A) and the projected panel area (PA).
It also means that each component, part of a LE device, will be define by only one of the three different parameters ($M_{COMP,}$ M_A or M_C)
This also explains how come the user input at the aircraft level (chord, span of LE devices, type of device) has a direct influence on the overall mass estimation.



	A	B	C	D	E	F	G	H	I	J	K
16	**Slat without slave tracks**										
17	**Elements**	**Number**		**mass/comp. (kg)**	**mass/area-length (kg**	**Panel chord (m)**	**Panel area (m2)**		**Mass (kg)**		
18	Slat	1			20.00		0		0.00		
19	Main tracks	2			10.00	0			0.00		
20	Main rollers/track	4		3.00					12.00		
21	Side rollers/track	4		2.00					8.00		
22	actuators/track	2		2.00					8.00		
23	Miscellaneous	1		3.00					3.00		
24								**TOTAL=**	**31.00 kg**		
25											
26											
27	**Fixed Camber Krueger flap**										
28	**Elements**	**Number**		**mass/comp. (kg)**	**mass/area-length (kg**	**Panel chord (m)**	**Panel area (m2)**		**Mass (kg)**		
29	Flap	1			20.00		0		0.00		
30	Mechanism	2			2.00	0			0.00		
31	Actuator/Mech	4		2.00					16.00		
32	bearings/mech.	2		2.00					8.00		
33	Miscellaneous	1		3.00					3.00		
34								**TOTAL=**	**27.00 kg**		
35											
36											
37											
38	**Fixed Camber Krueger flap with Folding nose**										
39	**Elements**	**Number**		**mass/comp. (kg)**	**mass/area-length (kg**	**Panel chord (m)**	**Panel area (m2)**		**Mass (kg)**		
40	Flap	1			20.00		0		0.00		
41	Actuators	3			10.00	0			0.00		
42	bearings/actuators	4		2.00					24.00		
43	Mec. Parts/actuator	2		2.00					12.00		
44	Fold. nose parts/actuators	4		1.00					12.00		
45	Miscellaneous	1		3.00					3.00		
46								**TOTAL=**	**51.00 kg**		
47											
48											
49											
50	**Variable camber Krueger flap**										
51	**Elements**	**Number**		**mass/comp. (kg)**	**mass/area-length (kg**	**Panel chord (m)**	**Panel area (m2)**		**Mass (kg)**		
52	Flap	1			20.00		0		0.00		
53	Mechanism	3			10.00	0			0.00		
54	Actuator/mech.	2		2.00					12.00		
55	Bearings/mech.	10		5.00					150.00		
56	Miscellaneous	1		3.00					3.00		
57								**TOTAL=**	**165.00 kg**		

Figure 87 - View of parameters for detail mass [Lajux]

Analytical description of the reliability estimation

The calculation used to obtain the reliability results are described below. To understand how the reliability tool generates results it is necessary to see the step by step calculations.

As the mass estimation tool, the reliability tool is structured using different steps from the "aircraft layer" (the overall result) to the "component layer" (the most detailed layer). Each of them uses specific equations, which are explained thereafter.

At the "aircraft level", the result generated is the overall reliability for the LE system on the aircraft:

$$F_{total} = 2 \times F_{wing}$$

Equation 8-7

F_{total}= Total failure rate for the whole aircraft LE system [per 10^6h]
F_{wing}= Total failure rate for the LE system of one wing [per 10^6h]

At the "wing level", this is the equation used to get reliability of one wing:

$$F_{wing} = F_{inb} + F_{central} + F_{outb}$$

Equation 8-8

F_{inb}= Total failure rate for inboard wing section [per 10^6h]
$F_{central}$=Total failure rate for central wing section [per 10^6h]
F_{outb}= Total failure rate for outboard wing section [per 10^6h]

At the "wing section level", this is the equation used to find the reliability of the inboard or outboard section of one wing:

$$F_s = \sum_{i}^{n} F_{Di}$$

Equation 8-9

F_S= Total failure rate for one wing section [per 10^6h]
F_D= Failure rate for one type of device [per 10^6h]

At the "device type level", it is possible to estimate the reliability for each type of device (this for many panels) on one section:

$$F_{Dd} = N_{Pd} + F_{Pd}$$

Equation 8-10

F_{Dd}=	Specific device type failure rate	[per 10^6h]
N_{Pd}=	Number of panel for each device/section	[--]
F_{Pd}=	Specific panel failure rate	[per 10^6h]

At the "panel level", the equation used to find the reliability is:

$$F_P = \sum_i^n F_{COMP,i}$$

Equation 8-11

F_P=	Panel failure rate	[per 10^6h]
F_{COMP}=	Component failure rate	[per 10^6h]

At the "component level", the equation gives the reliability for 1 or more component of the same kind:

$$F_{COMP} = \alpha(F_{COMP,cycle} + F_{COMP,time})$$

Equation 8-12

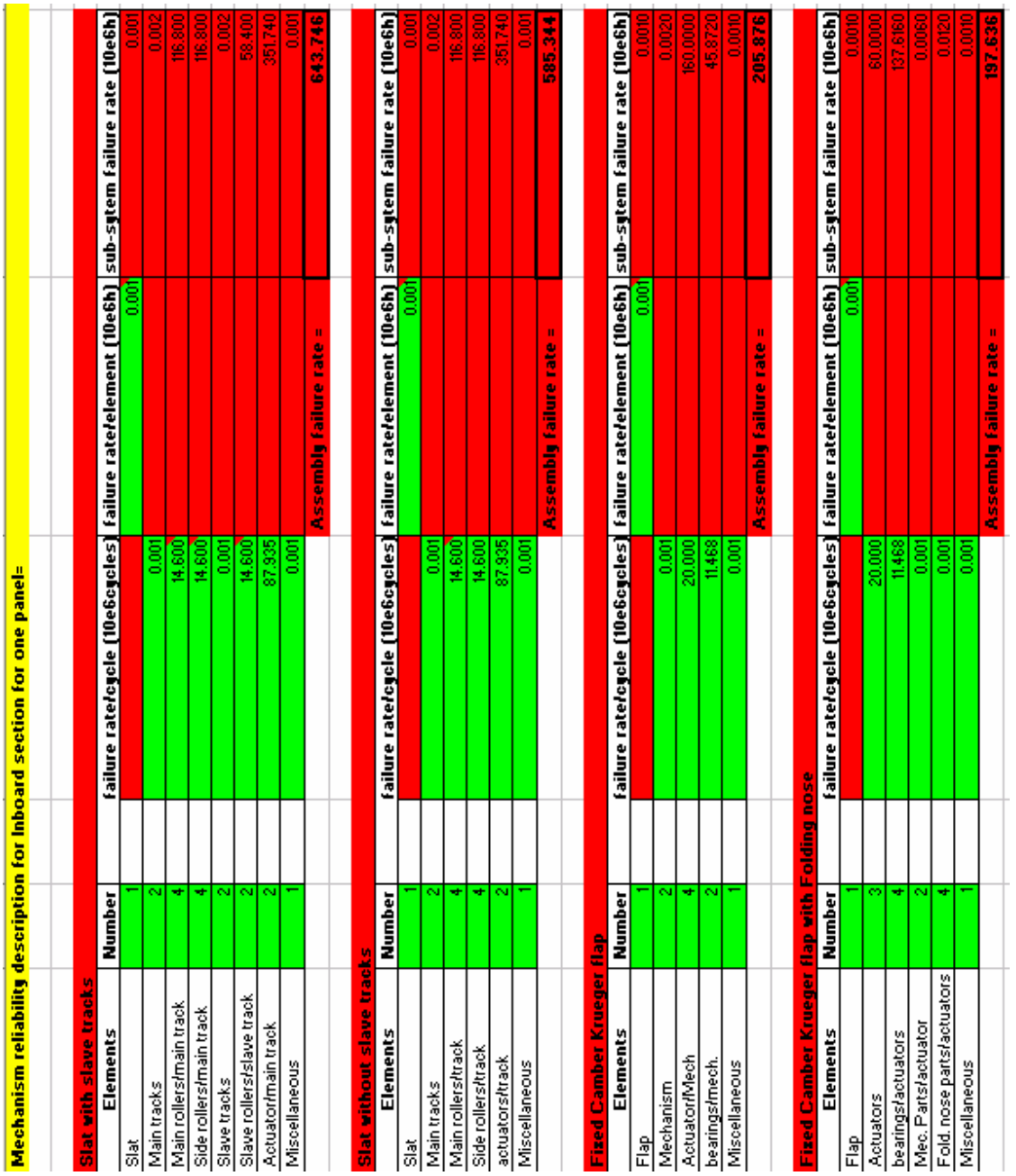
F_{COMP} =	Component failure rate	[per 10^6h]
α =	Number of component of each type	[--]
$F_{COMP,cycle}$ =	Specific component failure rate	[per 10^6cycle]
$F_{COMP,time}$ =	Specific component failure rate	[per 10^6h]

The specific component failure rate can be estimated by a number of cycle (for parts like actuators being used at take off and landing), or by a time (for parts such as flaps).

All of these equations are based on the fact that a system failure rate is equal to the summation of the sub-system failure rates as mentioned by Dr M. Bineid in his PhD thesis[35] .

$$\lambda_s = \sum_i^n \lambda_i$$

Equation 8-13



Mechanism reliability description for Inboard section for one panel=

Slat with slave tracks				
Elements	Number	failure rate/cycle (10e6cycles)	failure rate/element (10e6h)	sub-sytem failure rate (10e6h)
Slat	1		0.001	0.001
Main tracks	2	0.001		0.002
Main rollers/main track	4	14.600		116.800
Side rollers/main track	4	14.600		116.800
Slave tracks	2	0.001		0.002
Slave rollers/slave track	2	14.600		58.400
Actuator/main track	2	87.935		351.740
Miscellaneous	1	0.001		0.001
			Assembly failure rate =	643.746

Slat without slave tracks				
Elements	Number	failure rate/cycle (10e6cycles)	failure rate/element (10e6h)	sub-sytem failure rate (10e6h)
Slat	1		0.001	0.001
Main tracks	2	0.001		0.002
Main rollers/track	4	14.600		116.800
Side rollers/track	4	14.600		116.800
actuators/track	2	87.935		351.740
Miscellaneous	1	0.001		0.001
			Assembly failure rate =	585.344

Fixed Camber Krueger flap				
Elements	Number	failure rate/cycle (10e6cycles)	failure rate/element (10e6h)	sub-sytem failure rate (10e6h)
Flap	1		0.001	0.0010
Mechanism	2	0.001		0.0020
Actuator/Mech	4	20.000		160.0000
bearings/mech.	2	11.468		45.8720
Miscellaneous	1	0.001		0.0010
			Assembly failure rate =	205.876

Fixed Camber Krueger flap with Folding nose				
Elements	Number	failure rate/cycle (10e6cycles)	failure rate/element (10e6h)	sub-sytem failure rate (10e6h)
Flap	1		0.001	0.0010
Actuators	3	20.000		60.0000
bearings/actuators	4	11.468		137.6160
Mec. Parts/actuator	2	0.001		0.0060
Fold. nose parts/actuators	4	0.001		0.0120
Miscellaneous	1	0.001		0.0010
			Assembly failure rate =	197.636

Figure 88 - View of parameters for detailed reliability [Lajux]

Analytical description of the manufacturing cost estimation

The calculation used to obtain the manufacturing cost results are described below. To understand how this tool generates results it is necessary to see the step by step calculations.

As the mass estimation tool and the reliability tool, this is structured using different steps from the "aircraft layer" (the overall result) to the "component layer" (the most detailed layer). Each of them uses specific equations, which are explained thereafter.

The aircraft layer is keeping the same configuration as the other tools. So details on this can be found in the earlier tools description.

However, the detailed level used different types of parameters.

At the "component level", the equation to give the cost for the material to manufacture 1 or more component of the same kind is:

$$C_{MAT} = \alpha(M_{COMP} \times C_{SPEC} \times R_{B2F})$$

Equation 8-14

C_{MAT} =	Cost of material to buy to manufacture 1 type of component	[£]
α =	Number of component of each type	[--]
C_{SPEC} =	Specific cost to weight coefficient	[£/Kg]
$R_{B"F}$ =	"Buy to Fly" ratio	[--]

The cost of manufacturing each part is calculated through a flat manufacturing cost rate related to the overall size of the component to manufacture:

$$C_{Manuf} = R_{Manuf} \times D_{Comp}$$

Equation 8-15

C_{Manuf} =	Cost of manufacturing 1 type of component	[£]
R_{Manuf} =	Rate for manufacturing operations	[£/m]
D_{Comp} =	Dimension of components	[m]

The overall cost of manufacturing one type of parts (including when there are multiple of the same parts) is calculated by adding the cost of purchasing the material and the cost of manufacturing.

$$C_{comp} = C_{Manuf} + C_{MAT}$$

Equation 8-16

C_{comp} =	Overall cost of one type of component	[£]

The overall cost of an assembly is calculated by adding the cost of the different parts. The cost of assembly as been incorporated as part of the flat rate for manufacturing

$$C_{Assy} = \sum_{i}^{n} C_{Comp}$$

Equation 8-17

C_{Assy} = Overall cost of one type of assembly [£]

This cost of each assembly is then transferred to the aircraft level.

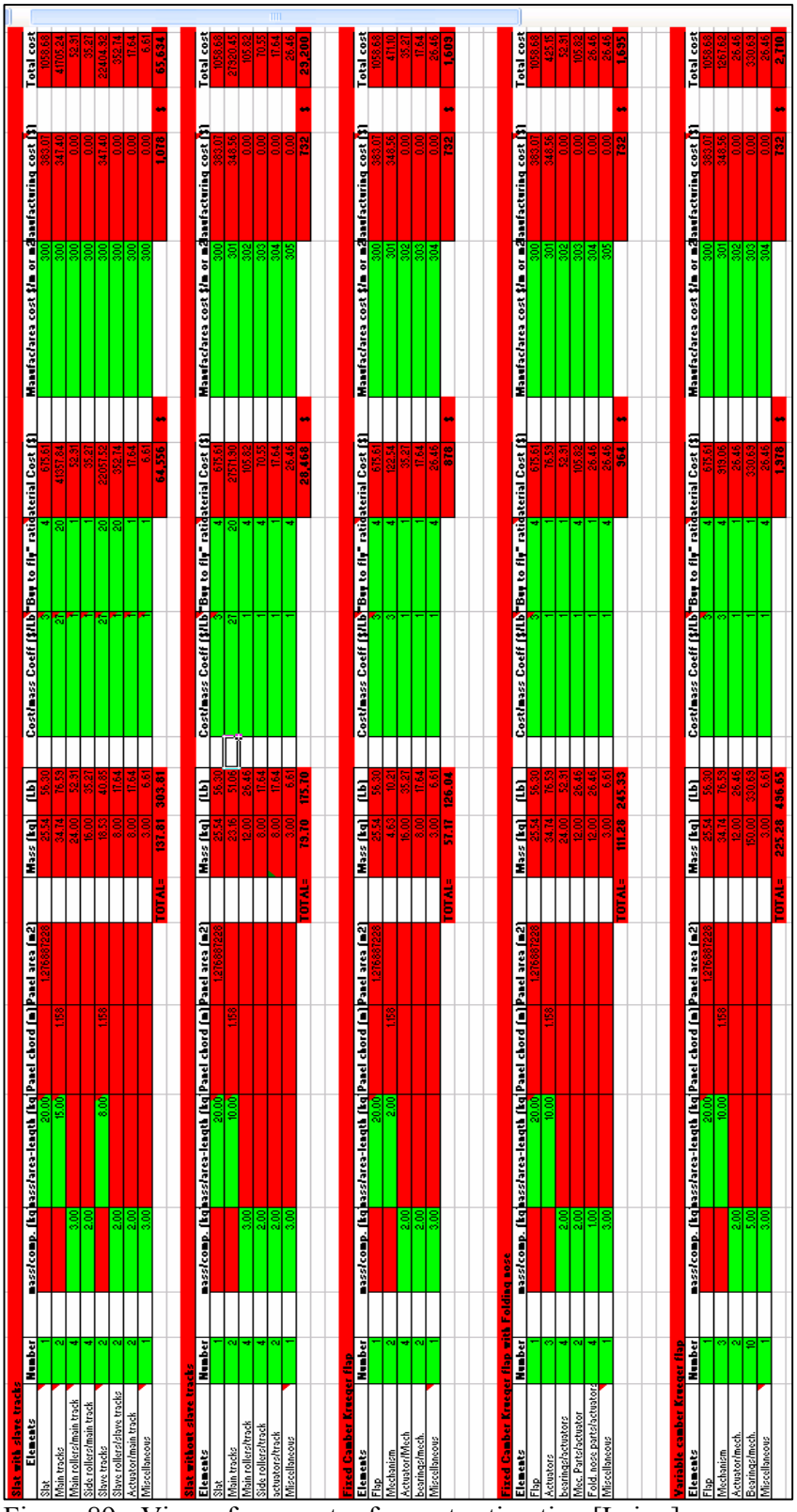


Figure 89 - View of parameter for cost estimation [Lajux]

Analytical description of the fleet management cost estimation

The calculations used to obtain the cost result (for the fleet management) are described below. To understand how this tool generates results it is necessary to see the step by step calculations.

The fleet management cost is based on the time to failure for on overall LE system for a whole aircraft; from there a full fleet cost management set of calculations has been developed.

The inputs are to be given by the users they are as follow=

The amount of flying hours per day = flying (h/day)
The amount of days flown during each year= flying (day/yea)
The number of years the aircraft is in service for = Year of use (years)
The average aircraft speed = A/C aver. Speed (km/h)
Average time for each flight for the different case = Short/ Medium / Long range (h)
Number of seats per A/C (seats)

Then the following values are created based on the multiplication of the different inputs to represent the right value:

Total flying time(h/year)
A/C. life flight time (h)
A/C life distance (km) or (miles)

From there the user can enter the number of aircraft in his fleet, and he also has to enter an average price value (£) for each average maintenance operations following failure (in £)

Following on this is possible to calculate the associated cost =

Average Monthly maintenance cost (£/ month (for one A/C))
Fleet Monthly maintenance cost (£/ month)
Fleet year maintenance cost (£)
Fleet lifetime cost (£M)
Average Maintenance cost per A/C (£/km)
Average Maintenance cost per A/C per seat (£/km.seat)

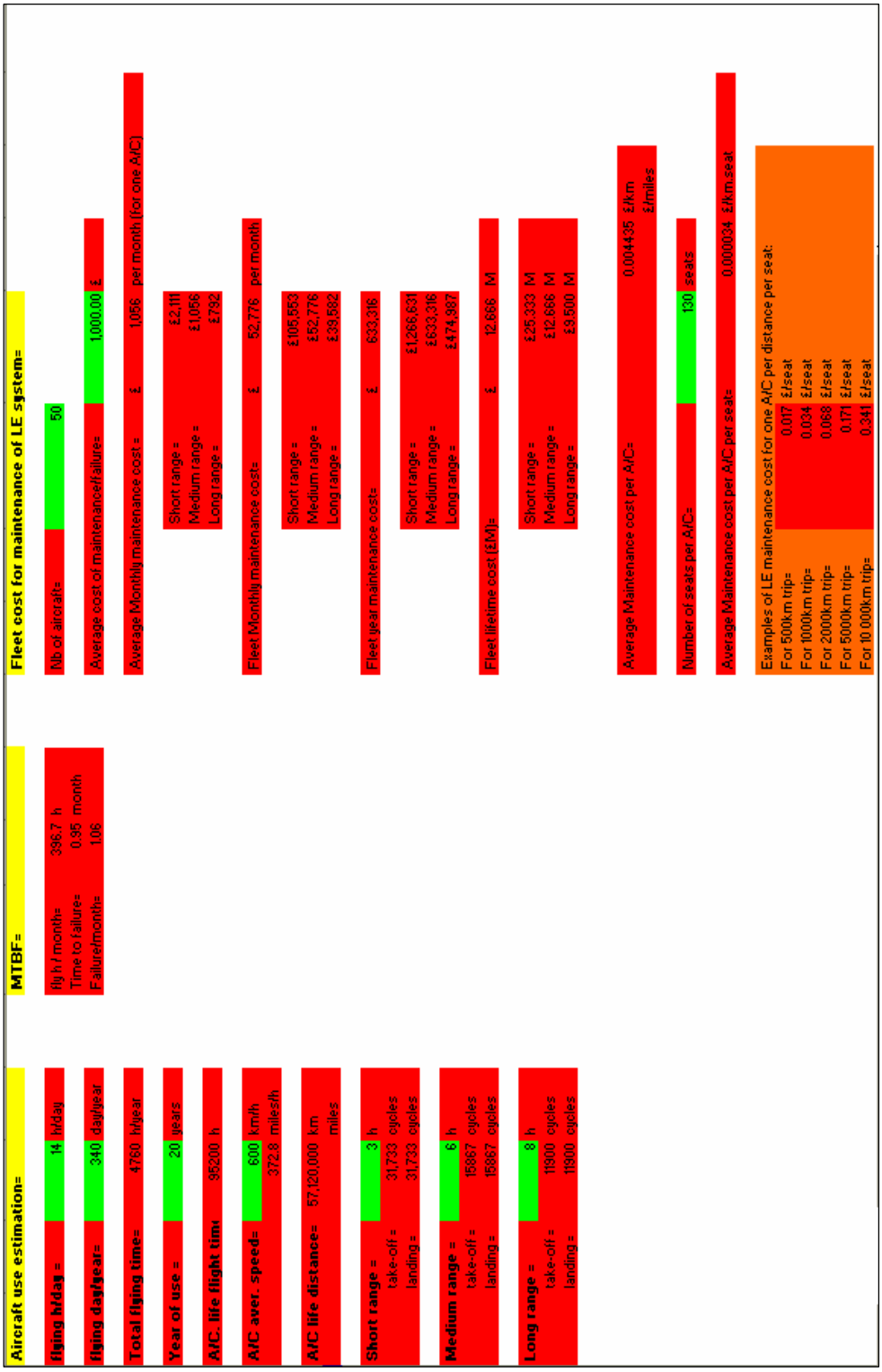


Figure 90 - detail of the fleet cost management tool [Lajux]

8.5. Appendixes D

Low speed case graphs:

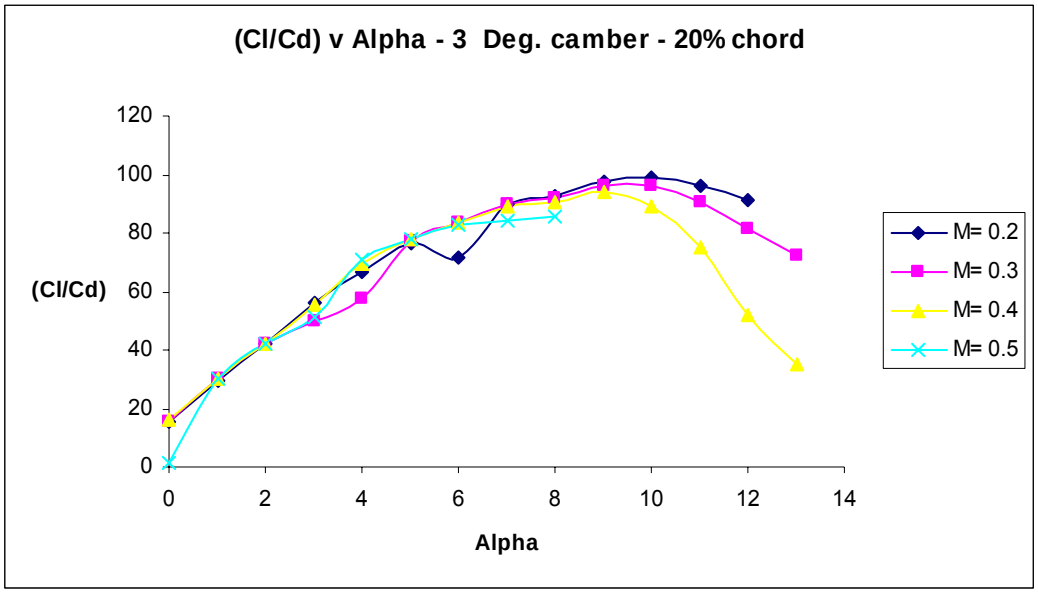


Figure 91 - L/D 3Dg camber [Lajux]

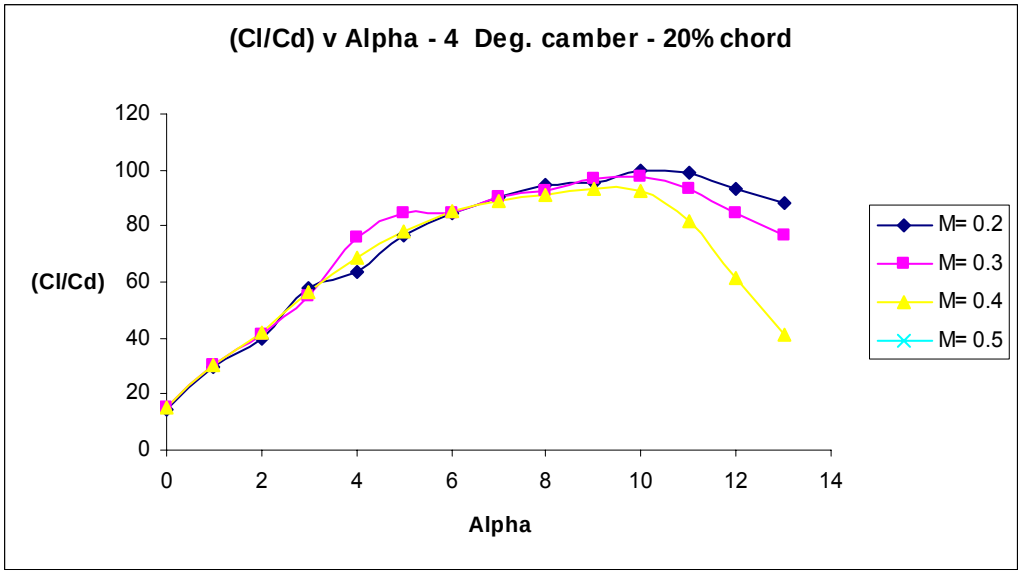


Figure 92 - L/D 4Dg camber [Lajux]

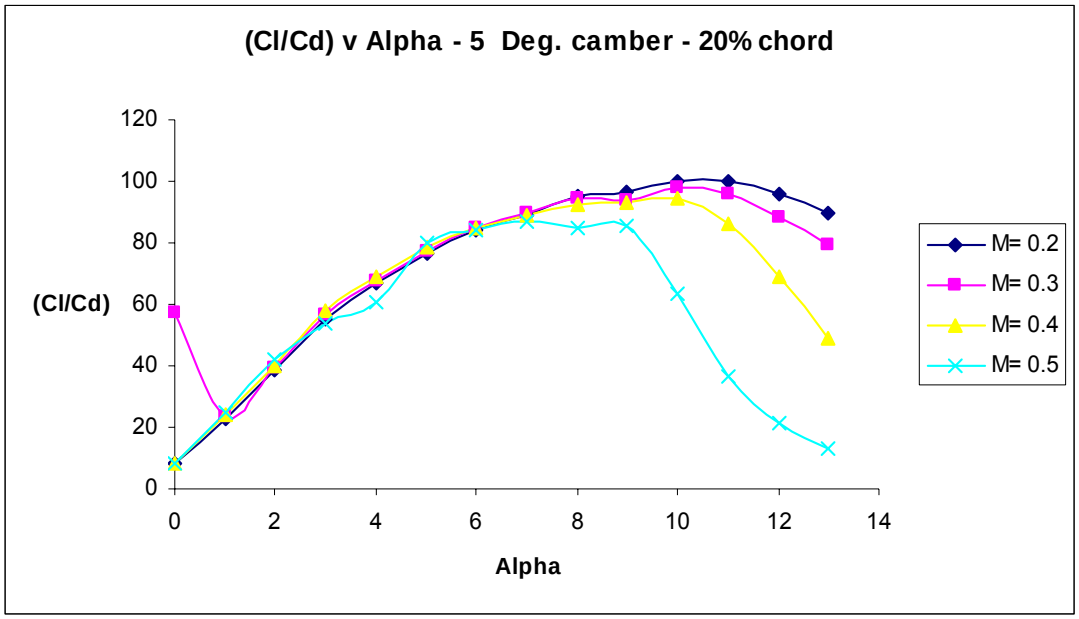


Figure 93 - L/D 5Dg camber [Lajux]

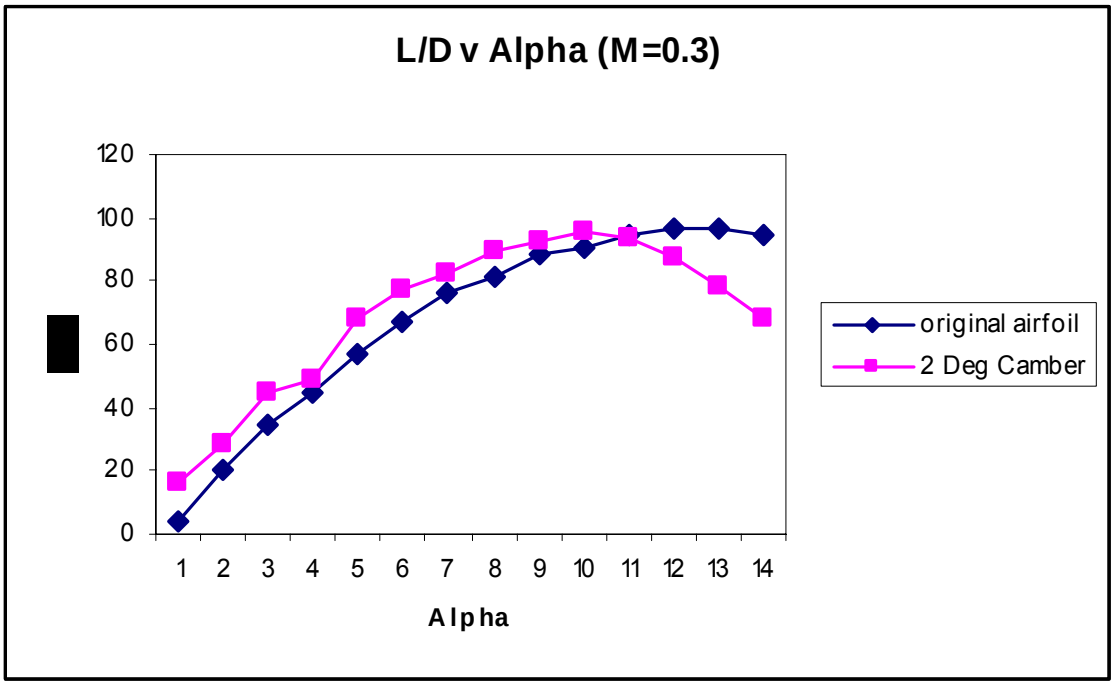


Figure 94 - L/D v alpha for 2Dg camber case [Lajux]

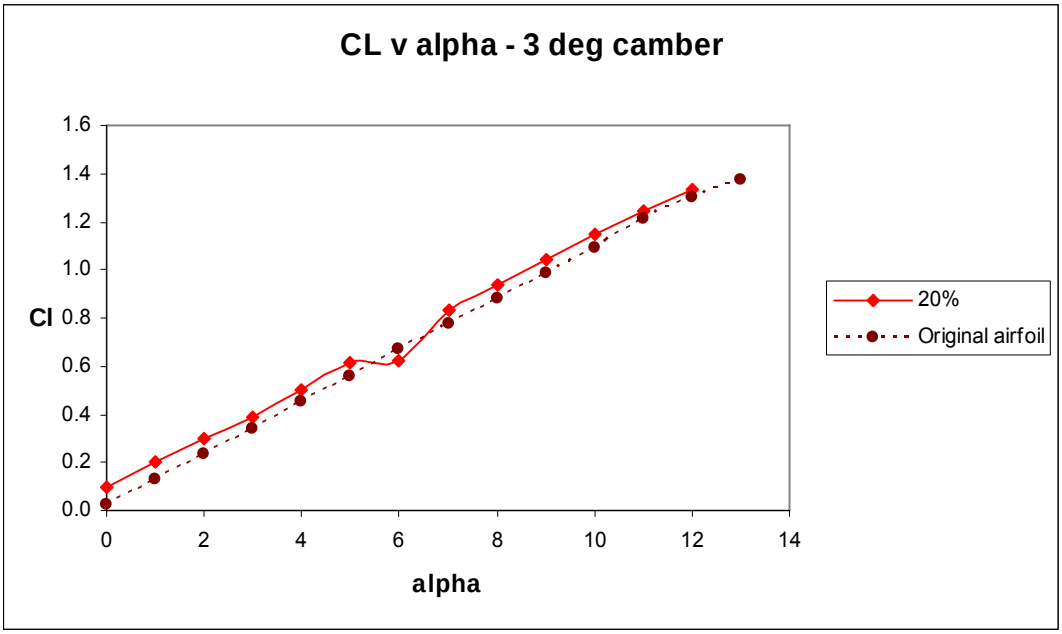


Figure 95 - Delta Cl: VC LE and original profile (M=0.3, 20% chord) [Lajux]

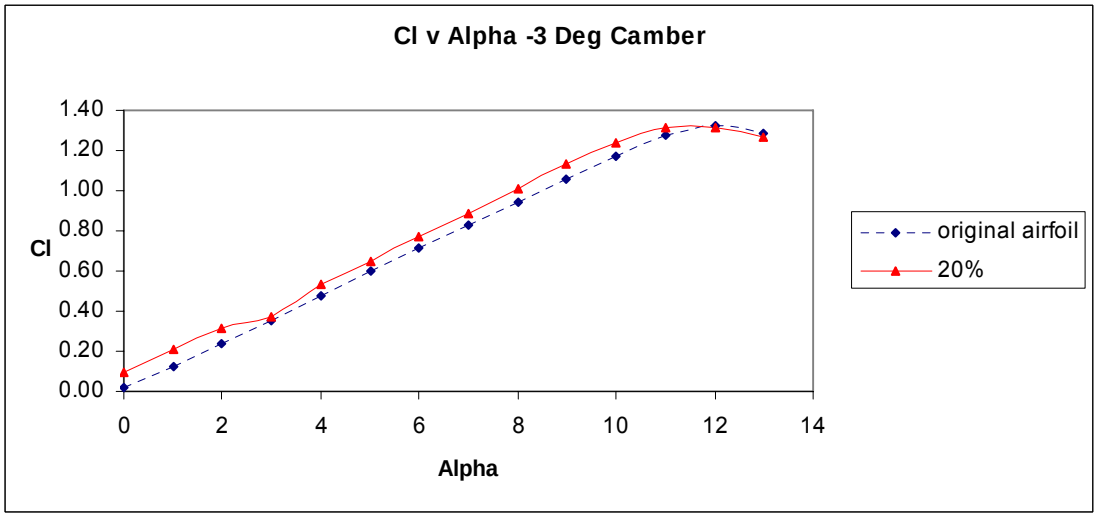


Figure 96 - Delta Cl: VC LE & original profile (M=0.4, 20% chord) [Lajux]

high speed case graphs:

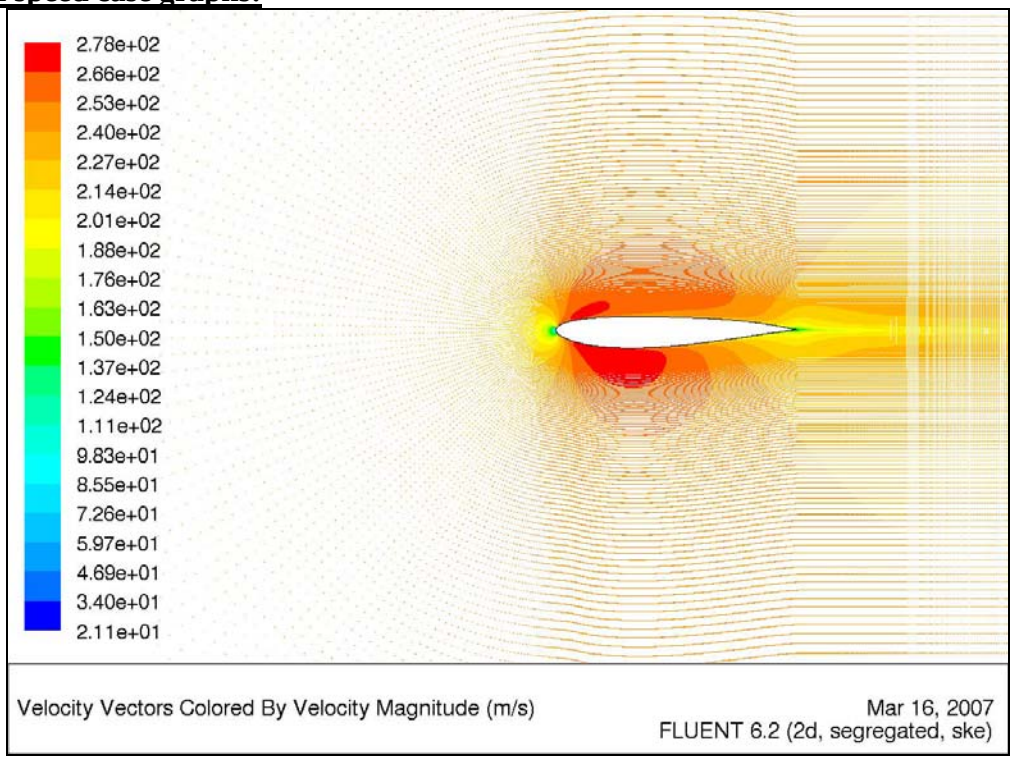


Figure 97 - Edi original - velocity vector [Lajux]

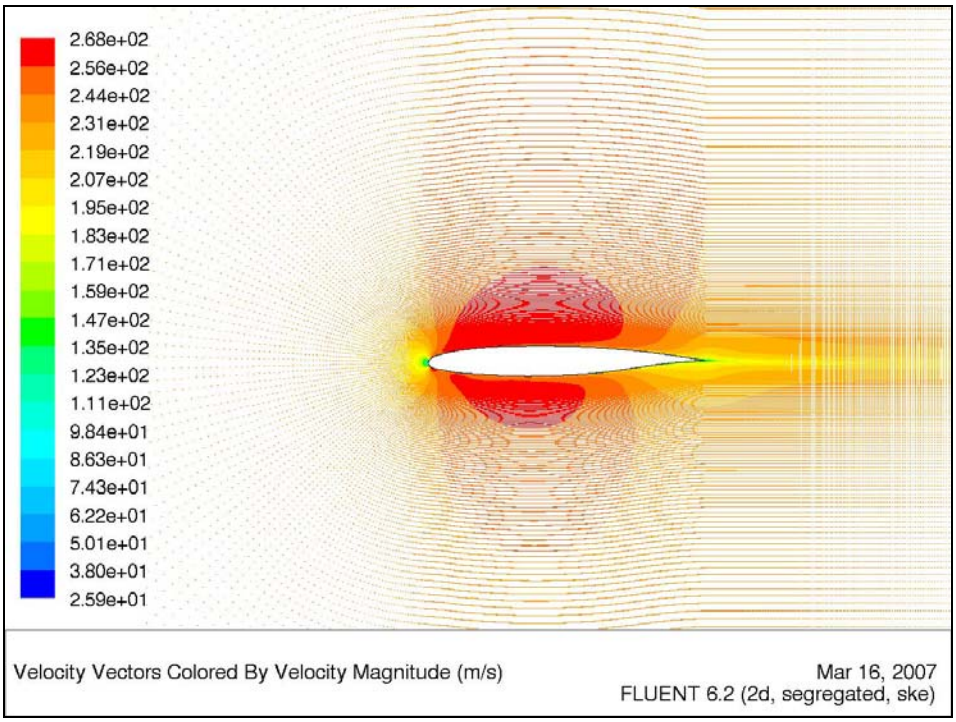


Figure 98 - Edi 2 Deg camber velocity vector

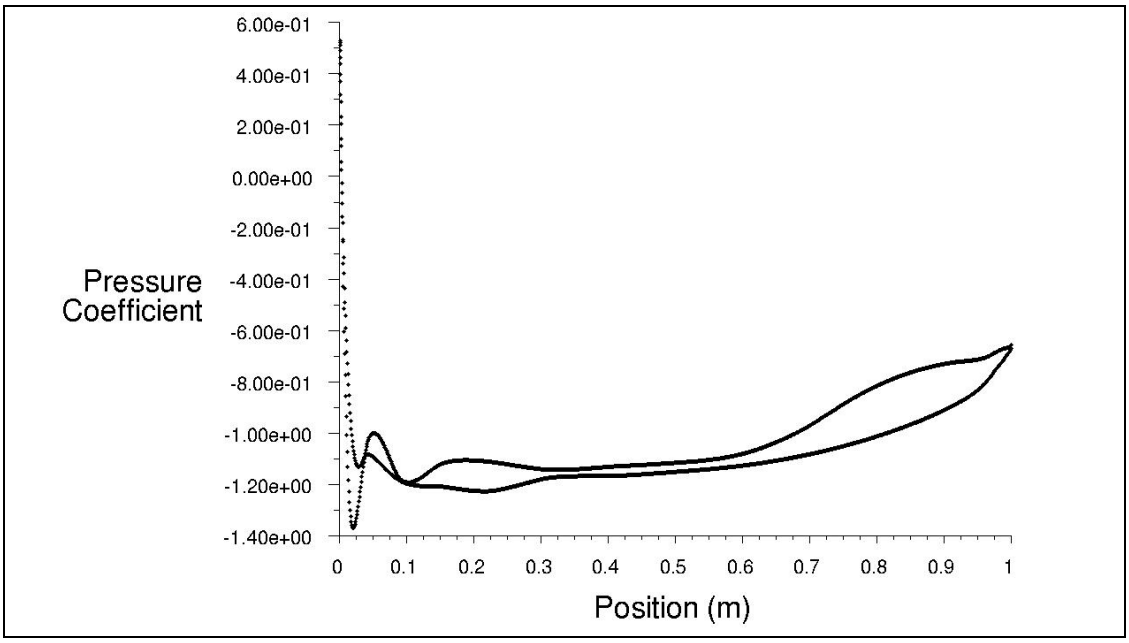


Figure 99 - Cp on a 2 Deg camber at high speed case [Lajux]

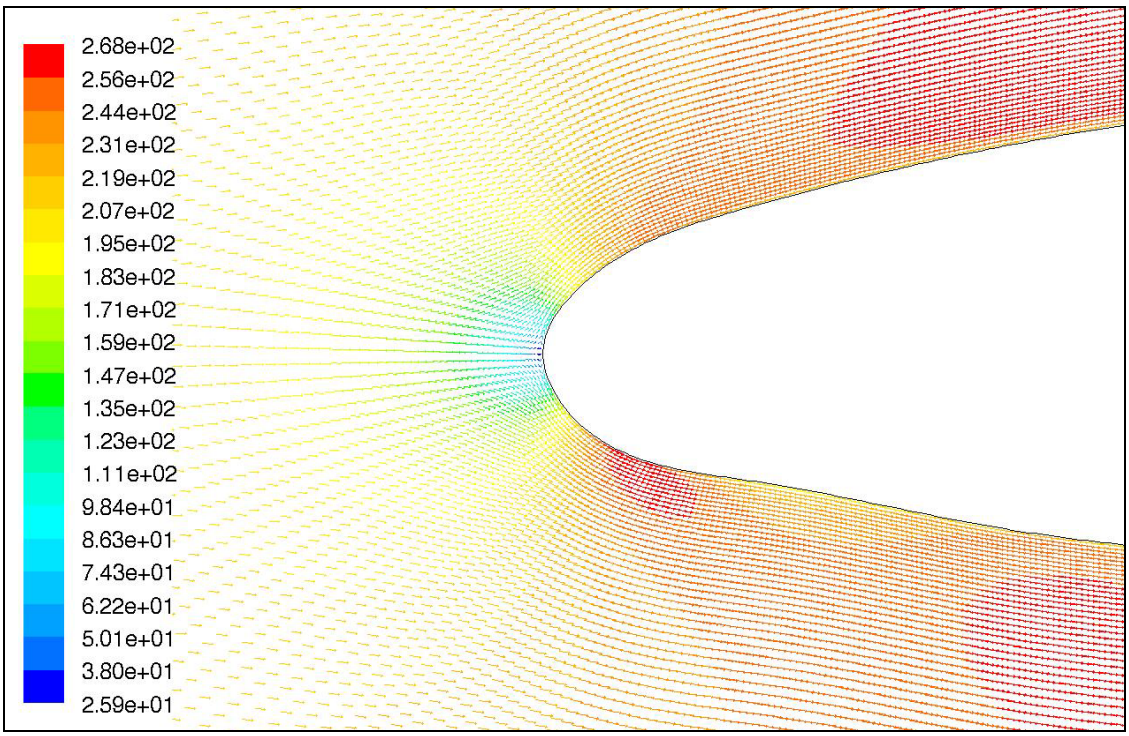


Figure 100 - Velocity vectors on 2 Deg. camber case [Lajux]