MASTER OF SCIENCE THESIS

Design of a Wing with Boundary Layer Suction

Redesigning the Wing of the Eaglet

H.J.B. van de Wal

August 24, 2010



Faculty of Aerospace Engineering · Delft University of Technology



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For obtaining the degree of Master of Science in Aerospace Engineering at Delft University of Technology

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August 24, 2010

Faculty of Aerospace Engineering · Delft University of Technology



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Delft University Of Technology Department Of Aerodynamics

The undersigned hereby certify that they have read and recommend to the Faculty of Aerospace Engineering for acceptance a thesis entitled "Design of a Wing with Boundary Layer Suction" by H.J.B. van de Wal in partial fulfillment of the requirements for the degree of Master of Science.

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Abstract

Over the last century lots of efforts have been put in the reduction of profile drag. By using advanced techniques in airfoil design, the passive ways to reduce profile drag by shaping have come to a standstill due to physical limits. To further reduce the profile drag, an active method has to be used.

Boundary layer suction is one of these active methods and its effect is twofold. A laminar boundary layer will be stabilized, preventing transition and yielding larger areas of laminar flow, which generate less drag. On the other hand, turbulent boundary layer separation will be postponed, resulting in a higher maximum lift coefficient.

In this thesis it is investigated how much improvement can be achieved by implementing boundary layer suction on the Euro-ENAER EE–10 Eaglet, a research aircraft of the Delft University of Technology. For this aircraft, a new airfoil has been designed in XFOIL which is optimized for boundary layer suction. The new airfoil proved to have good aerodynamic properties with and without suction and showed vast improvements in profile drag. Additionally, the maximum lift coefficient is increased significantly.

Also the effects of boundary layer suction on flap and aileron deflection have been investigated. Results showed a significant decrease in drag and increase in maximum lift.

With this newly designed airfoil, a new wing was created and its aerodynamic properties were calculated using the lifting line implementation of XFLR. The new wing proved to be more efficient, a drag reduction of 13% was achieved at cruise up to 20% at high flight speeds. However, the drag reduction of the total aircraft was marginally due to the high drag of the rest of the aircraft. At cruise the drag reduction of the total aircraft was about 3.2%.

Acknowledgements

First of all I wish to thank my family. Throughout my studies they always supported me unconditionally and gave me the advice and strength to keep going.

Also I want to thank my girlfriend, for being so patient and giving me the love and support I needed.

I would like to thank my friends for a wonderful time here in Delft.

Finally I wish to thank my colleagues and mr. Boermans for their help and guidance during the final phase of my study.

Delft, The Netherlands August 24, 2010 Bart van de Wal

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Nomenclature

Latin Symbols

| A | Aspect ratio | [-] |
|-----------|--|--|
| a | Speed of sound | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| b | Wing span | [-] |
| c | Chord | [m] |
| C_1 | Arbitrary constant | [-] |
| C_1 | Coefficient in Diederich's method | [-] |
| C_2 | Arbitrary constant | [-] |
| C_2 | Coefficient in Diederich's method | [-] |
| C_3 | Arbitrary constant | [-] |
| C_3 | Coefficient in Diederich's method | [-] |
| C_4 | Arbitrary constant | [-] |
| C_4 | Coefficient in Diederich's method | [-] |
| C_D | Three-dimensional drag coefficient | [-] |
| c_d | Two-dimensional drag coefficient | [-] |
| C_f | Skin-friction coefficient | [-] |
| c_g | Mean geometric chord | [m] |
| C_L | Three-dimensional lift coefficient | [-] |
| c_l | Two-dimensional lift coefficient | [-] |
| c_v | Specific heat at constant volume | $\left[\frac{J}{kg\cdot K}\right]$ |
| C_{D_0} | Three-dimensional drag at $C_L = 0$ | [-] |
| C_{L_0} | Three-dimensional lift coefficient at $\alpha = 0$ | [-] |

| $C_{L_{\alpha}}$ | Three-dimensional lift-curve slope | [-] |
|------------------|--|---|
| $c_{l_{lpha}}$ | Two-dimensional lift-curve slope | [-] |
| c_{la} | Additional lift distribution | [-] |
| c_{l_b} | Basic lift distribution | [-] |
| E | Jone's edge-velocity correction | [-] |
| e | Internal energy per unit mass | $\left[\frac{J}{kg}\right]$ |
| F | Planform parameter in Diederich's method | [-] |
| f | Correction factor for wing taper | [-] |
| Н | Shape factor | [-] |
| i | Incidence angle of the wing | [-] |
| k | Correction factor for chord ratio of split flaps | [-] |
| k | Thermal conductivity | $\left[\frac{\text{kg} \cdot \text{m}}{s^2 K}\right]$ |
| L | Three-dimensional lift force | [N] |
| l | Two-dimensional lift force | $\left[\frac{N}{m}\right]$ |
| L_a | Anderson's additional lift function | [-] |
| L_b | Anderson's basic lift function | [-] |
| M | Mach number | [-] |
| N | Amplification factor | [-] |
| $O(\ldots)$ | Order of magnitude | |
| p | Pressure | [Pa] |
| q | Volumetric heat per unit mass | $\left[\frac{J}{kg}\right]$ |
| R | Specific gas constant | $\left[\frac{J}{kg \cdot K}\right]$ |
| r | Magnitude of vector \vec{r} | [m] |
| Re | Reynolds number | [-] |
| Re_{θ} | Reynolds number with respect to the momentum thickness | [-] |
| Re_c | Reynolds number with respect to the chord | [-] |
| S | Area | $[m^2]$ |
| s | Path coordinate | [m] |
| SCC | Shear lag factor | [-] |
| T | Temperature | [K] |
| t | Time | [s] |
| U | Absolute velocity at the boundary layer edge | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| u | Velocity component in x -direction | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| V | Absolute velocity | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| v | Induced-drag factor | [-] |
| v | Velocity component in y -direction | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| w | Induced-drag factor | [-] |
| w | Velocity component in z-direction | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |

| x | Cartesian coordinate | [m] |
|---|----------------------|-----|
| y | Cartesian coordinate | [m] |
| z | Cartesian coordinate | [m] |

Greek Symbols

| α | Angle of attack | [-] |
|------------------|---|--|
| α_{0_1} | Local aerodynamic twist at the spanwise station for which $c_{l_b} = 0$ | [-] |
| $lpha_{l_0}$ | Zero-lift angle of an airfoil section | [-] |
| β | Prandtl-Glauert compressibility correction | [-] |
| δ | Boundary layer thickness | [m] |
| δ | Control-surface or flap deflection | [-] |
| δ | Increment of plane wing induced drag coefficient due to additional lift | [-] |
| $\Delta(\ldots)$ | Difference in $()$ | |
| δ^* | Displacement thickness | [m] |
| ϵ | Aerodynamic twist | [-] |
| η | Non-dimensional spanwise station | [-] |
| Г | Circulation | $\left[\frac{1}{s}\right]$ |
| Г | Dihedral of the wing | [-] |
| γ | Ratio of specific heats | [-] |
| γ | Vortex | $\left[\frac{\mathrm{m}}{\mathrm{s}}\right]$ |
| Λ | Sweep angle | [-] |
| λ | Bulk viscosity coefficient | $\left[\frac{\mathrm{kg}}{\mathrm{m}\cdot\mathrm{s}}\right]$ |
| λ | Taper ratio | [-] |
| Λ_eta | Corrected sweep angle | [-] |
| μ | Dynamic viscosity coefficient | $\left[\frac{\mathrm{kg}}{\mathrm{m}\cdot\mathrm{s}}\right]$ |
| ν | Kinematic viscosity coefficient | $\left[\frac{\mathrm{m}^2}{\mathrm{s}}\right]$ |
| Ψ | Stream function | $\left[\frac{m^2}{s}\right]$ |
| ρ | Density | $\left[\frac{\mathrm{kg}}{\mathrm{m}^3}\right]$ |
| σ | Source | $\left[\frac{\mathrm{m}^2}{\mathrm{s}}\right]$ |
| au | Shear stress | [Pa] |
| θ | Angle of vector \vec{r} | [-] |
| θ | Momentum thickness | [m] |
| ξ | Streamwise coordinate | [-] |

Subscripts

| 0 | At the wall | |
|---------------|---|--|
| ϵ | Twist | |
| $\frac{c}{4}$ | At quart chord | |
| ∞ | At infinity | |
| crit | Critical | |
| c | Cruise | |
| e | At the boundary layer edge | |
| f | Flap | |
| h | Horizontal tailplane | |
| i | Induced | |
| p | Profile | |
| r | Root | |
| t | Tip | |
| v | Vertical tailplane | |
| w | Wing | |
| xx | In the plane of x in the direction of x | |
| xy | In the plane of x in the direction of y | |
| xz | In the plane of x in the direction of z | |
| yx | In the plane of y in the direction of x | |
| yy | In the plane of y in the direction of y | |
| yz | In the plane of y in the direction of z | |
| zx | In the plane of z in the direction of x | |
| zy | In the plane of z in the direction of y | |
| <i>zz</i> | In the plane of z in the direction of z | |

Superscripts

| () | Flux $\left(\frac{\mathrm{d}}{\mathrm{d}t}\right)$ |
|-----|--|
| ()′ | Dimensionless |
| () | Vector |

Abbreviations

| \mathbf{AC} | Aerodynamic Center |
|---------------|--------------------|
| \mathbf{AC} | Angle of Climb |

| ENAER | Empresa Nacional de Aeronáutica de Chile |
|----------------|---|
| ISA | International Standard Atmosphere |
| JAR | Joint Aviation Requirements |
| \mathbf{LE} | Leading Edge |
| \mathbf{LLT} | Lifting Line Theory |
| MAC | Mean Aerodynamic Chord |
| MGC | Mean Geometric Chord |
| NACA | National Advisory Committee for Aeronautics |
| OAT | Outside Air Temperature |
| RC | Rate of Climb |
| RPM | Revolutions Per Minute |
| SHP | Shaft Horsepower |
| VFR | Visual Flight Rules |

Other Symbols

- \downarrow Control surface deflection down
- $\leftarrow \qquad \qquad \text{Control surface deflection left}$
- \rightarrow Control surface deflection right
- $\uparrow \qquad \qquad {\rm Control\ surface\ deflection\ up}$

Chapter 1

Introduction

1.1 About the Eaglet

In June 1986, the Chilean aircraft manufacturer ENAER (Empresa Nacional de Aeronáutica de Chile) started a project, named Avión Liviano, to design a small aircraft. They designed an aircraft called Ñamcu, which means eaglet in Mapudungun. The Ñamcu is a two-seat, composite light aircraft. Construction of the first prototype (CC–PZI), see Figure 1.1, started in February 1987 and its maiden flight was in April 1989 (Jackson [1996]).



Figure 1.1: The ENAER Ñamcu CC-PZI in flight (Jackson [1996])

The Namcu was only certified for flying in Chile, and for that reason could not be sold in other countries. In September 1995, a new founded joint venture in The Netherlands, called Euro-ENAER, supported by the Faculty of Aerospace Engineering of the Delft University of Technology, started with the JAR 23 certification of the Namcu. The name of the JAR 23 certified ENAER Namcu was changed into the Euro-ENAER Eaglet. Production of the Eaglet was delayed due to problems with its certification, and although the problems were resolved in 2001, the company was announced bankrupt in January 2002 (Jackson [2002]).

1.2 About the Thesis

The Faculty of Aerospace Engineering of the Delft University of Technology bought one of the Eaglets, the PH–EAG, see Figure 1.2, when Euro-ENAER went bankrupt.



Figure 1.2: The Euro-ENAER Eaglet PH–EAG in flight (Delft University of Technology [2009])

This aircraft is mainly used for research. A new field of research for the Eaglet is the reduction of drag by means of boundary layer suction. Boundary layer suction is a technique, where a part of the laminar boundary layer is sucked away to maintain its stability, preventing it to transit in a turbulent one and preventing a turbulent boundary layer to separate. Research has already been done on the Eaglet by Debrauwer [2008]. He applied boundary layer suction on the existing wing of the Eaglet. The goal of this thesis is:

Designing a New Wing for the Eaglet, Optimized for Boundary Layer Suction.

The design process is focused on the reduction of drag during cruise conditions, while keeping the general planform of the wing constant.

1.3 About Boundary Layer Suction

Boundary layer suction is a technique, where part of the boundary layer is sucked away through the wing (or other body), see Figure 1.3. There are two reasons to apply boundary layer suction; one is to postpone separation, the other reason is to postpone transition. To postpone separation, a part of the turbulent boundary layer is sucked away, which will prevent the growth of the boundary layer and keeping it attached, therefore preventing separation. In doing so, it is possible to fly with higher angles of attack and lower velocities.



Figure 1.3: Schematic representation of boundary layer suction (Boermans [2008])

To postpone transition of the boundary layer from a laminar one to a turbulent one and avoid it to separate, laminar boundary layer suction can be used. A portion of the laminar boundary layer will be removed which stabilizes the boundary layer. This is because the growth of instabilities in the laminar boundary layer will decrease, i.e. the Tollmien-Schlichting waves¹ will be damped. Because this results in larger areas of laminar flow, the profile drag will be reduced.

The reason that a laminar boundary layer gives lower profile drag compared to a turbulent one is twofold; the friction drag as well as the pressure drag is reduced. Figure 1.4 shows the velocity profiles of a laminar and a turbulent boundary layer. Clearly can be seen that the turbulent one is 'fuller' than the laminar one. Its velocity gradient is larger, which will result in a larger friction:

$$\tau = \mu \left(\frac{\partial u}{\partial y}\right)_0$$

Besides the lower friction, a laminar boundary layer also has a lower pressure drag. In Figure 1.5 the boundary layer development from laminar to turbulent can be seen. After transition the boundary layer thickness grows, causing a higher pressure drag; the flow outside the boundary layer has to be displaced over a larger thickness, giving a resisting force.

¹Tollmien-Schlichting waves are disturbances in the boundary layer which will amplify due to the noslip condition at the wall. When these waves grow beyond a certain value, they will trigger boundary layer transition.



Figure 1.4: Laminar velocity profile versus a turbulent one (Anderson Jr. [2001])



Figure 1.5: Velocity profile of a laminar and turbulent boundary layer (Anderson Jr. [2001])

1.4 Outline of the Thesis

The outline of this thesis is as follows. In Chapter 2 the suction equations are derived, starting with the Navier-Stokes equations. Also some important quantities for boundary layers are defined. Chapter 3 discusses the program XFOIL, its versions and modifications, and doing batch-analysis in XFOIL. Also the program XFLR is discussed. Followed by Chapter 4, where the characteristics of the original Eaglet and its wing are calculated. In this chapter a handbook method and the program XFLR are used to calculate the aerodynamic properties of the original wing. Chapter 5 discusses the design procedure of the new airfoil, starting with the requirements followed by the new design and its suction distribution. With this new airfoil, a new wing is created, as described in Chapter 6. The suction requirements are discussed, followed by the spanwise suction distribution. Also presented is the selection of the flaps and the effects of suction on aileron deflections. Chapter 7 presents the final design. It discusses the chosen design and shows the improvements achieved by the new wing with boundary layer suction. Finally, in Chapter 8 conclusions are drawn and recommendations are given for the design and further investigations.

Chapter 2

Theoretical Background

2.1 The General Case: Navier-Stokes

An unsteady, compressible, three-dimensional viscous flow is described by the Navier-Stokes equations. Strictly speaking, the Navier-Stokes equations are the three components of the momentum equation (2.2), (2.3) and (2.4). However, to close the problem they also contain the continuity equation (2.1) and the energy equation (2.5). The energy equation in its turn can be closed by the thermodynamic relation $e = c_v T$ and the equation of state for a perfect gas $p = \rho RT$. The only assumptions made in the derivation of the Navier-Stokes equations are that the fluid is Newtonian and the flow is a non-reacting continuum (Anderson Jr. [2001]).

The continuity equation is given by:

$$\frac{\partial \rho}{\partial t} + \frac{\partial (\rho u)}{\partial x} + \frac{\partial (\rho v)}{\partial y} + \frac{\partial (\rho w)}{\partial z} = 0$$
(2.1)

The three components of the momentum equation are:

$$\rho \frac{\partial u}{\partial t} + \rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} + \rho w \frac{\partial u}{\partial z} = -\frac{\partial p}{\partial x} + \frac{\partial}{\partial x} \left(\lambda \nabla \cdot \vec{V} + 2\mu \frac{\partial u}{\partial x} \right) + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \right] + \frac{\partial}{\partial z} \left[\mu \left(\frac{\partial u}{\partial z} + \frac{\partial w}{\partial x} \right) \right]$$
(2.2)

$$\rho \frac{\partial v}{\partial t} + \rho u \frac{\partial v}{\partial x} + \rho v \frac{\partial v}{\partial y} + \rho w \frac{\partial u}{\partial z} = -\frac{\partial p}{\partial y} + \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \right] + \frac{\partial}{\partial y} \left(\lambda \nabla \cdot \vec{V} + 2\mu \frac{\partial v}{\partial y} \right) + \frac{\partial}{\partial z} \left[\mu \left(\frac{\partial w}{\partial y} + \frac{\partial v}{\partial z} \right) \right]$$
(2.3)

$$\rho \frac{\partial w}{\partial t} + \rho u \frac{\partial w}{\partial x} + \rho v \frac{\partial w}{\partial y} + \rho w \frac{\partial w}{\partial z} = -\frac{\partial p}{\partial z} + \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial u}{\partial z} + \frac{\partial w}{\partial x} \right) \right] \\ + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial w}{\partial y} + \frac{\partial v}{\partial z} \right) \right] + \frac{\partial}{\partial z} \left(\lambda \nabla \cdot \vec{V} + 2\mu \frac{\partial w}{\partial z} \right)$$
(2.4)

And finally the energy equation:

$$\rho \frac{\partial \left(e + \frac{V^2}{2}\right)}{\partial t} + \rho u \frac{\partial \left(e + \frac{V^2}{2}\right)}{\partial x} + \rho v \frac{\partial \left(e + \frac{V^2}{2}\right)}{\partial y} + \rho w \frac{\partial \left(e + \frac{V^2}{2}\right)}{\partial z}$$

$$= \rho \dot{q} + \frac{\partial}{\partial x} \left(k \frac{\partial T}{\partial x}\right) + \frac{\partial}{\partial y} \left(k \frac{\partial T}{\partial y}\right) + \frac{\partial}{\partial z} \left(k \frac{\partial T}{\partial z}\right)$$

$$- \nabla \cdot p \vec{V} + \frac{\partial (u\tau_{xx})}{\partial x} + \frac{\partial (u\tau_{yx})}{\partial y} + \frac{\partial (u\tau_{zx})}{\partial z}$$

$$+ \frac{\partial (v\tau_{xy})}{\partial x} + \frac{\partial (v\tau_{yy})}{\partial y} + \frac{\partial (v\tau_{zy})}{\partial z}$$

$$+ \frac{\partial (w\tau_{xz})}{\partial x} + \frac{\partial (w\tau_{yz})}{\partial y} + \frac{\partial (w\tau_{zz})}{\partial z}$$
(2.5)

where:

$$\tau_{xy} = \tau_{yx} = \mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \quad \tau_{xx} = \lambda (\nabla \cdot \vec{V}) + 2\mu \frac{\partial u}{\partial x}$$

$$\tau_{yz} = \tau_{zy} = \mu \left(\frac{\partial w}{\partial y} + \frac{\partial v}{\partial z} \right) \quad \tau_{yy} = \lambda (\nabla \cdot \vec{V}) + 2\mu \frac{\partial v}{\partial y}$$

$$\tau_{zx} = \tau_{xz} = \mu \left(\frac{\partial u}{\partial z} + \frac{\partial w}{\partial x} \right) \quad \tau_{zz} = \lambda (\nabla \cdot \vec{V}) + 2\mu \frac{\partial w}{\partial y}$$

2.2 First Assumptions

The Navier-Stokes equations are highly non-linear partial differential equations, and in their most general form, Equations (2.1) to (2.5), have no analytical solutions. To obtain analytical solutions, simplifications have to be made. These simplifications depend on the problem at hand.

The Eaglet is a low-subsonic aircraft, and flies in conditions where compressibility effects are ignorable. Also, during cruising flight and steady climb or descend, the flight conditions are assumed to change marginally. Therefore steady conditions can be assumed. Furthermore, when investigating boundary layers, an infinite wing is assumed, therefore the flow phenomena in one dimension can be ignored. Also the curvature of the bodies is assumed to be small, with smooth transitions. These assumptions imply that the density (ρ) can be treated as a constant, that the energy equation (2.5) is redundant and the unsteady terms $\left(\frac{\partial}{\partial t}\right)$ can be ignored. Also the third dimension and the second order derivatives in x will vanish due to the two-dimensional analysis and the small curvature.

The continuity equation (2.1) becomes:

$$\rho \frac{\partial u}{\partial x} + \rho \frac{\partial v}{\partial y} = 0$$
$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0$$
$$\nabla \cdot \vec{V} = 0$$
(2.6)

Using this result (2.6) and the above assumptions for the x and y components of the momentum equation, Equations (2.2) and (2.3) will simplify into:

$$\rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} = -\frac{\partial p}{\partial x} + \frac{\partial}{\partial y} \left[\mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \right]$$
(2.7)

$$\rho u \frac{\partial v}{\partial x} + \rho v \frac{\partial v}{\partial y} = -\frac{\partial p}{\partial y} + \frac{\partial}{\partial x} \left[\mu \left(\frac{\partial v}{\partial x} + \frac{\partial u}{\partial y} \right) \right]$$
(2.8)

2.3 Dimensionless Navier-Stokes

The Navier-Stokes equations, derived in Section 2.2, can be made dimensionless by introducing the following variables:

$$u' = \frac{u}{U_{\infty}} \quad v' = \frac{v}{U_{\infty}} \quad x' = \frac{x}{c} \quad y' = \frac{y}{c}$$

$$\rho' = \frac{\rho}{\rho_{\infty}} \quad p' = \frac{p}{p_{\infty}} \quad \mu' = \frac{\mu}{\mu_{\infty}}$$
(2.9)

where U_{∞} , ρ_{∞} , p_{∞} and μ_{∞} are the reference values (e.g. freestream values) and c is a reference length (e.g. the chord length). Replacing the variables in the continuity equation (2.6), with dimensionless ones, will result in the dimensionless continuity equation:

$$\frac{U_{\infty}}{c}\frac{\partial u'}{\partial x'} + \frac{U_{\infty}}{c}\frac{\partial v'}{\partial y'} = 0$$

$$\frac{\partial u'}{\partial x'} + \frac{\partial v'}{\partial y'} = 0$$
(2.10)

The same can be done with the components of the momentum equation. Inserting variables (2.9) into Equation (2.7):

$$\frac{\rho_{\infty}U_{\infty}U_{\infty}}{c}\rho'u'\frac{\partial u'}{\partial x'} + \frac{\rho_{\infty}U_{\infty}U_{\infty}}{c}\rho'v'\frac{\partial u'}{\partial y'} = -\frac{p_{\infty}}{c}\frac{\partial p'}{\partial x'} + \frac{\mu_{\infty}U_{\infty}}{c \cdot c}\frac{\partial}{\partial y'}\left[\mu'\left(\frac{\partial v'}{\partial x'} + \frac{\partial u'}{\partial y'}\right)\right]$$
(2.11)

Rewriting the factors:

$$\frac{p_{\infty}}{\rho_{\infty}U_{\infty}^2} = \frac{\gamma p_{\infty}}{\gamma \rho_{\infty}U_{\infty}^2} = \frac{a_{\infty}^2}{\gamma U_{\infty}^2} = \frac{1}{\gamma M_{\infty}^2} \qquad \text{and} \\ \frac{\mu_{\infty}}{\rho_{\infty}U_{\infty}c} = \frac{1}{\text{Re}_{\infty}},$$

and inserting them into Equation (2.11), gives the dimensionless x-momentum equation:

$$\rho' u' \frac{\partial u'}{\partial x'} + \rho' v' \frac{\partial u'}{\partial y'} = -\frac{1}{\gamma M_{\infty}^2} \frac{\partial p'}{\partial x'} + \frac{1}{\text{Re}_{\infty}} \frac{\partial}{\partial y'} \left[\mu' \left(\frac{\partial v'}{\partial x'} + \frac{\partial u'}{\partial y'} \right) \right]$$
(2.12)

Following the same procedure for the y-momentum equation (2.8), will yield the dimensionless y-momentum equation:

$$\rho' u' \frac{\partial v'}{\partial x'} + \rho' v' \frac{\partial v'}{\partial y'} = -\frac{1}{\gamma M_{\infty}^2} \frac{\partial p'}{\partial y'} + \frac{1}{\text{Re}_{\infty}} \frac{\partial}{\partial x'} \left[\mu' \left(\frac{\partial v'}{\partial x'} + \frac{\partial u'}{\partial y'} \right) \right]$$
(2.13)

Equations (2.10), (2.12) and (2.13) are the starting point for the derivation of the boundary layer equations.

2.4 The Boundary Layer Equations

In 1904, Prandtl presented his concept of the boundary layer (Anderson Jr. [2005]). He subdivided the flow over a body into two distinct regions. An outer region where the flow is assumed inviscid, and a region close to the surface, the boundary layer, where viscous effects are dominating, see Figure 2.1.

In general, the boundary layer is very thin compared to the dimensions of the body on which it flows. This causes large velocity gradients in the boundary layer and, according to the proportionality of the velocity gradient to the shear stress in Newton's shear-stress law, contributes to a non-ignorable skin-friction.

With the concept of the boundary layer, the assumption can be made that the boundary layer thickness δ is small compared to the body-length c. In mathematical form:

 $\delta \ll c$

With this result, an order of magnitude analysis can be made to determine which terms in the momentum equation (2.7) and (2.8) are small compared to others and can be ignored.


Figure 2.1: Subdivision of the flow. An inviscid outer flow and a viscous boundary layer. (The Free Dictionary [2009])

From the dimensionless variables (2.9) can be seen that u' varies from 0 at the surface, to 1 at the edge of the boundary layer. Therefore it can be said that u' has an order of magnitude of 1, O(1). Because variable x varies from 0 to c, x' also has an order of magnitude of 1, O(1). The same holds for ρ' , p' and μ' . However, y varies from 0 to δ , and $\delta \ll c$, resulting that y' has a smaller order of magnitude, $y' = O(\delta)$. The order of magnitude of v' is still unknown. Substituting the orders of magnitude in the continuity equation (2.10) yields:

$$\frac{O(1)}{O(1)} + \frac{v'}{O(\delta)} = 0 \tag{2.14}$$

Variable v' must have an order of magnitude of δ , $O(\delta)$ for Equation (2.14) to be correct. Substituting the obtained results into the x-momentum equation (2.12) gives:

$$O(1) + O(1) = -\frac{1}{\gamma M_{\infty}^2} O(1) + \frac{1}{\text{Re}_{\infty}} \left[O(1) + O\left(\frac{1}{\delta^2}\right) \right]$$
(2.15)

Making the assumption that the Reynolds number is large, in the order of δ^{-2} , Equation (2.15) becomes:

$$O(1) + O(1) = -\frac{1}{\gamma M_{\infty}^2} O(1) + O(\delta^2) \left[O(1) + O\left(\frac{1}{\delta^2}\right) \right]$$
(2.16)

The term $O(\delta^2)[O(1)]$, corresponding to $\frac{1}{\text{Re}_{\infty}} \frac{\partial}{\partial y'} \left(\mu' \frac{\partial v'}{\partial x'} \right)$, is much smaller in order of magnitude than the other terms in Equation (2.16), and can therefore be neglected:

$$\rho' u' \frac{\partial u'}{\partial x'} + \rho' v' \frac{\partial u'}{\partial y'} = -\frac{1}{\gamma M_{\infty}^2} \frac{\partial p'}{\partial x'} + \frac{1}{\operatorname{Re}_{\infty}} \frac{\partial}{\partial y'} \left(\mu' \frac{\partial u'}{\partial y'} \right)$$
(2.17)

In dimensional form Equation (2.17) becomes:

$$\rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} = -\frac{\partial p}{\partial x} + \frac{\partial}{\partial y} \left(\mu \frac{\partial u}{\partial y} \right)$$

Doing the same analysis for the y-momentum equation (2.13) yields:

$$O(\delta) + O(\delta) = -\frac{1}{\gamma M_{\infty}^2} \frac{\partial p'}{\partial y'} + O(\delta^2) \left[O(\delta) + O\left(\frac{1}{\delta}\right) \right]$$
(2.18)

All the terms in Equation (2.18) are of order of magnitude of $O(\delta)$, or smaller. Assuming that $\gamma M_{\infty}^2 = O(1)$ follows that $\frac{\partial p'}{\partial y'}$ must be $O(\delta)$ or smaller. The *y*-momentum equation (2.8) reduces to:

$$\frac{\partial p}{\partial y} = 0 \tag{2.19}$$

Equation (2.19) tells that p is not a function of y and therefore only a function of x, p = p(x). Finally it is assumed that μ also is a function of x only. The set of boundary layer equations in their final form are given by:

continuity:
$$\frac{\partial u}{\partial x} + \frac{\partial v}{\partial y} = 0$$
 (2.20)

x-momentum:
$$\rho u \frac{\partial u}{\partial x} + \rho v \frac{\partial u}{\partial y} = -\frac{\mathrm{d}p}{\mathrm{d}x} + \mu \frac{\partial^2 u}{\partial y^2}$$
 (2.21)

y-momentum:
$$\frac{\partial p}{\partial y} = 0$$
 (2.22)

2.5 Boundary Layer Suction (1)

The boundary layer equations (2.20) to (2.22) are the starting point for the derivation of the boundary layer suction equation. Starting with Prandtl's equation (2.21):

$$u\frac{\partial u}{\partial x} + v\frac{\partial u}{\partial y} = -\frac{1}{\rho}\frac{\mathrm{d}p}{\mathrm{d}x} + \nu\frac{\partial^2 u}{\partial y^2} \qquad \text{with: } \nu = \frac{\mu}{\rho}$$
(2.23)

and substituting the following boundary conditions:

$$u(x,0) = 0$$
 $v(x,0) = v_0(x)$
 $u(x,\infty) = U(x)$

will yield Equation (2.24):

$$v_0 \left(\frac{\partial u}{\partial y}\right)_0 = -\frac{1}{\rho} \frac{\mathrm{d}p}{\mathrm{d}x} + \nu \left(\frac{\partial^2 u}{\partial y^2}\right)_0 \tag{2.24}$$

Euler's equation is given by:

$$\mathrm{d}p = -\rho u \,\mathrm{d}u$$

Applying this to the boundary layer edge $u(x, \infty)$, will result in:

$$-\frac{1}{\rho}\frac{\mathrm{d}p}{\mathrm{d}x} = U\frac{\mathrm{d}U}{\mathrm{d}x}$$

and substituting this into Equation (2.24), yields:

$$v_0 \left(\frac{\partial u}{\partial y}\right)_0 = U \frac{\mathrm{d}U}{\mathrm{d}x} + \nu \left(\frac{\partial^2 u}{\partial y^2}\right)_0$$

Assuming a flat-plate boundary layer, which has a linear velocity distribution near the surface, in other words $\left(\frac{\partial^2 u}{\partial y^2}\right)_0 = 0$, results into the boundary layer suction equation:

$$v_0 \left(\frac{\partial u}{\partial y}\right)_0 = U \frac{\mathrm{d}U}{\mathrm{d}x} \tag{2.25}$$

It states that the suction velocity depends on the boundary layer edge velocity and its first derivative in x.

2.6 Important Quantities in Boundary Layers

To transform the boundary layer suction equation (2.25) into a more convenient form, some quantities have to be defined.

2.6.1 The Displacement Thickness

When comparing a viscous flow situation to an inviscid one, it can be seen that there is a mass-flow deficit because the boundary layer is retarding the flow, see Figure 2.2. This mass-flow deficit causes the free flow to divert, as if the body had a different shape, see Figure 2.3. This phenomena is modeled by the displacement thickness δ^* .

The displacement thickness can be derived by calculating the difference in mass-flow between the inviscid and viscous situation. The mass-flow for the inviscid situation is given by:

$$\int_{0}^{\tilde{y}} \rho U dy \qquad \tilde{y} \to \infty \tag{2.26}$$

The mass-flow for the viscous situation:

$$\int_{0}^{\tilde{y}} \rho u dy \qquad \tilde{y} \to \infty \tag{2.27}$$



Figure 2.2: Inviscid situation versus the viscous situation (Anderson Jr. [2001])



Figure 2.3: Boundary layer displacement (Anderson Jr. [2001])

The missing mass-flow due to the boundary layer $\rho U \delta^*$, is the difference in mass-flow between the inviscid (2.26) and the viscous (2.27) situation:

$$\rho U \delta^* = \int_0^{\tilde{y}} \rho U dy - \int_0^{\tilde{y}} \rho u dy$$
$$\rho U \delta^* = \int_0^{\tilde{y}} (\rho U - \rho u) dy$$
$$\Rightarrow \delta^* = \int_0^{\tilde{y}} \left(1 - \frac{u}{U}\right) dy \qquad \tilde{y} \to \infty$$
(2.28)

The displacement thickness is a measure for the pressure drag.

2.6.2 The Momentum Thickness

The momentum thickness is, similar to the displacement thickness, the deficit in momentum due to the boundary layer. Momentum is defined as the mass-flow multiplied by the velocity. In the definition of the momentum thickness, the mass-flow is assumed to be the mass-flow of the viscous situation. Therefore the momentum of the inviscid situation is:

$$\int_{0}^{\tilde{y}} U\rho u dy \qquad \tilde{y} \to \infty \tag{2.29}$$

And the momentum of the viscous situation is:

$$\int_{0}^{\tilde{y}} u\rho u dy \qquad \tilde{y} \to \infty \tag{2.30}$$

The difference between Equations (2.29) and (2.30) is the momentum deficit $\rho U^2 \theta$:

$$\rho U^{2} \theta = \int_{0}^{\tilde{y}} U \rho u dy - \int_{0}^{\tilde{y}} u \rho u dy$$

$$\rho U^{2} \theta = \int_{0}^{\tilde{y}} \rho u (U - u) dy$$

$$\Rightarrow \theta = \int_{0}^{\tilde{y}} \frac{u}{U} \left(1 - \frac{u}{U} \right) dy \qquad \tilde{y} \to \infty$$
(2.31)

The momentum thickness is a measure for the sum of the pressure drag and the skinfriction drag. A graphical representation of momentum and displacement thickness can be seen in Figure 2.4.



Figure 2.4: Momentum and displacement thickness (White [2006])

Table 2.1: The shape factor for a laminar boundary layer

| Flow condition | Н |
|------------------|------|
| Stagnation point | 2.2 |
| Pressure minimum | 2.6 |
| Separation point | 4 |
| Separated flow | > 10 |

2.6.3 The Shape Factor

The shape factor is defined by the quotient of the displacement thickness and the momentum thickness:

$$H = \frac{\delta^*}{\theta} \tag{2.32}$$

It represents the condition of the flow, i.e. the state in which the boundary layer is. Typical conditions are stagnation, pressure minimum and separation. Some values of shape factors, typical for laminar boundary layers are shown in Table 2.1. The velocity profiles at different shape factors can be seen in Figure 2.5.

2.7 Boundary Layer Suction (2)

Now the above quantities are defined, the derivation of the boundary layer suction equations can be continued. It is convenient to derive a non-dimensional form. Multiplying the left hand side of Equation (2.25) with $\frac{U}{\theta} \frac{\theta}{U}$, which are functions of x only, we get:

$$v_0 \frac{U}{\theta} \left(\frac{\partial \left(\frac{u}{U} \right)}{\partial \left(\frac{y}{\theta} \right)} \right)_0 = U \frac{\mathrm{d}U}{\mathrm{d}x} \tag{2.33}$$

Blasius [1908] calculated for a flat-plate laminar boundary layer that:

$$\left(\frac{\partial(\frac{u}{U})}{\partial(\frac{y}{\theta})}\right)_0 = 0.2205$$



Figure 2.5: Velocity profiles of a laminar boundary layer at different shape factors: At s_1 pressure minimum (H = 2.6), at s_2 separation point (H = 4), and at s_3 separated flow (H > 10) (Anderson Jr. [2001])

Using this result, and multiplying the right hand side of Equation (2.33) with $\frac{U_{\infty}}{c} \frac{c}{U_{\infty}}$, we obtain:

$$0.2205v_0\frac{U}{\theta} = U\frac{U_\infty}{c}\frac{\partial\left(\frac{U}{U_\infty}\right)}{\partial\left(\frac{x}{c}\right)}$$

Rewriting and dividing by U_{∞} yields:

$$\frac{v_0}{U_{\infty}} = \frac{1}{0.2205} \frac{U\theta}{U_{\infty}c} \frac{U_{\infty}}{U} \frac{\mathrm{d}\left(\frac{U}{U_{\infty}}\right)}{\mathrm{d}\left(\frac{x}{c}\right)}$$
(2.34)

Defining:

$$\operatorname{Re}_{\theta} = \frac{U\theta}{\nu}$$
 and $\operatorname{Re}_{c} = \frac{U_{\infty}c}{\nu}$

and substituting this into Equation 2.34 will yield:

$$\frac{v_0}{U_{\infty}} = \frac{1}{0.2205} \frac{1}{\frac{U}{U_{\infty}}} \frac{\mathrm{d}\left(\frac{U}{U_{\infty}}\right)}{\mathrm{d}\left(\frac{x}{c}\right)} \frac{\mathrm{Re}_{\theta}}{\mathrm{Re}_{c}},\tag{2.35}$$

Practical results show that Re_{θ} is almost constant. Hence, for a first step in the iteration process, Re_{θ} be treated as a constant and equal to the value calculated at the start of the suction region.

Equation 2.35 gives the suction velocity distribution as a function of the boundary layer edge velocity and its first derivative in x.

Chapter 3

XFOIL and XFLR

3.1 Introduction to XFOIL

XFOIL is an interactive program for the design and analysis of subsonic isolated airfoils (Drela [2001]). It is written by Mark Drela in 1986 and updated until April 2008, when the last version, 6.97, was released. XFOIL is able to design and do viscous analysis of airfoils. It is able to calculate free transition or impose forced transition, handle transitional separation bubbles, cope with limited trailing edge separation, calculate lift and drag just beyond $c_{l_{max}}$ and use the Karman-Tsien compressibility correction. In XFOIL it is possible to generate NACA 4 and 5-digit airfoils, and has various options to modify the geometry of imported airfoils.

However, the main strength of XFOIL is that it is able to calculated a geometry out of a pressure distribution, provided that it is physically possible (Full-Inverse method). Also, it has an option to change only part of the pressure distribution, keeping the shape of the rest of the airfoil unchanged (Mixed-Inverse method). XFOIL uses a high-order panel method with fully coupled viscous/inviscid interaction. Its methodology is described in Drela [1989].

3.1.1 Inviscid Formulation

XFOIL uses for its inviscid formulation a simple linear-vorticity stream function panel method. A two-dimensional inviscid flowfield is constructed by the superposition of a freestream flow, a vortex sheet of strength γ and a source sheet of strength σ on the airfoil surface, see Figure 3.1. The trailing edge thickness is modeled by a source panel.

The stream function is given by:

$$\Psi(x,y) = u_{\infty}y - v_{\infty}x + \frac{1}{2\pi}\int\gamma(s)\ln r(s;x,y)\,\mathrm{d}s + \frac{1}{2\pi}\int\sigma(s)\theta(s;x,y)\,\mathrm{d}s$$



Figure 3.1: Airfoil and wake paneling in XFOIL

where r is the magnitude of the vector between the path coordinate s and field point x, y and θ the vector's angle. To close the equations, an explicit Kutta condition is implied.

The source sheet σ can be altered in the iteration process, to model the displacement effect of the boundary layer. To compensate for compressibility effects, a Karman-Tsien compressibility correction is used. This gives good results until high subsonic conditions, where the theory of Karman-Tsien fails and the accuracy rapidly degrades.

3.1.2 Inverse Formulation

Two inverse methods can be used in XOIL, the Full-Inverse method and the Mixed-Inverse method. The Full-Inverse method uses Lighthill's and Van Ingen's complex mapping method. It calculates the whole airfoil geometry out of entire surface speed distribution. The Mixed-Inverse method is the inviscid panel formulation, however instead of the panel vortex strengths being the unknowns, the panel node coordinates are the unknowns where ever the surface speed is described. This means that only a part of the airfoil is changed, making it able to alter only a specific section of the airfoil.

3.1.3 Viscous Formulation

Both the boundary layer and the wake are calculated with a two-equation integral boundary layer formulation, Equation (3.1) and Equation (3.2), and an envelope e^n transition criterion.

$$\frac{\mathrm{d}\theta}{\mathrm{d}\xi} + \left(2 + H - M_e^2\right)\frac{\theta}{u_e}\frac{\mathrm{d}u_e}{\mathrm{d}\xi} = \frac{C_f}{2} + \left\{\frac{v_0}{u_e}\right\}$$
(3.1)

$$\theta \frac{\mathrm{d}H^*}{\mathrm{d}\xi} + (2H^{**} + H^*(1-H))\frac{\theta}{u_e}\frac{\mathrm{d}u_e}{\mathrm{d}\xi} = 2C_D - H^*\frac{C_f}{2} + \left\{(1-H^*)\frac{v_0}{u_e}\right\}$$
(3.2)

In these equations, the terms between curly brackets account for boundary layer suction and were added by Ferreira [2002], see Section 3.2.1. The original Drela code does not include these terms. Shape factors H^* and H^{**} are defined respectively:

$$H^* = \frac{\int_0^\infty \frac{u}{\overline{U}} \left(1 - \left(\frac{u}{\overline{U}}\right)^2\right) \,\mathrm{d}y}{\int_0^\infty \frac{u}{\overline{U}} \left(1 - \left(\frac{u}{\overline{U}}\right)\right) \,\mathrm{d}y}$$

$$H^{**} = \left(\frac{0.064}{H_k - 0.8} + 0.251\right) M_e^2 \qquad \text{with} \qquad H_k = \frac{H - 0.290 M_e^2}{1 + 0.113 M_e^2}$$

The whole viscous solution has a strong interaction with the incompressible potential one, by using the surface transpiration model. Transpiration is a technique in which extra non-physical normal flows are created on an airfoil surface in order to form a new streamline pattern such that the surface streamlines no longer follow the airfoil surface under inviscid flow (Yiu & Stow [2005]). This enables calculating regions of limited separated flow.

The total velocity on all locations of the airfoil surface and its wake, the airfoil's surface vorticity distribution and the equivalent viscous source distribution, is calculated with the panel methods with the Karman-Tsien compressibility correction.

3.2 Modifications of XFOIL

At the Low Speed Laboratory of the Delft University of Technology, XFOIL is modified to improve its accuracy, to make it able to calculate the suction distribution for a constant shapefactor boundary layer and to calculate the effects of boundary layer suction.

3.2.1 Ferreira

Ferreira [2002] implemented suction in XFOIL. He added the VDES menu, where it was possible to create and manually modify suction distributions. It also had the option to load and save, scale and modify suction distributions, but was very basic and not yet user-friendly. He also implemented the transition prediction method of Van Ingen [1956]. The standard method for transition prediction (the 'Drela method'), is unable to calculate the damping of Tollmien-Schlichting waves. When the boundary layer encounters a favorable pressure gradient ($\frac{\partial p}{\partial x} < 0$) or in the case of boundary layer suction, the Tollmien-Schlichting waves are damped, moving the transition point aft and lowering the amplification factor N. The Drela transition method however, cannot handle this behavior and will keep the amplification factor constant until it starts growing again, see Figure 3.2¹. It can be seen that after 45% chord the amplification factor is constant, indicating that the Drela method fails.

The Van Ingen method implemented by Ferreira can handle this damping of Tollmien-Schlichting waves, as shown in Figure 3.3. It can be seen that the amplification factor decreases, indicating the damping of Tollmien-Schlichting waves. However, when the shape factor of Figure 3.3 is decreased, also the implemented Van Ingen method of Ferreira fails, as shown in Figure 3.4. This is because this implementation of the Van Ingen method keeps the amplification factor constant whenever $\text{Re}_{\theta} < \text{Re}_{\theta_{crit}}$.

and

¹In these figures, the x-axis is read in the following way. From left to right, starting left with the trailing edge, via the lower airfoil surface to the leading edge at the center of the figure, followed by the upper airfoil surface leading to the trailing edge at the right. In this thesis only suction is applied to the upper surface. Therefore, the right hands side of the figures is important with respect to boundary layer suction.



Figure 3.2: Drela transition method fails to calculate the amplification factor for H = 2.44 correctly. Displayed is the calculated suction distribution for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$ and $c_l = 0.45$.



Figure 3.3: Van Ingen transition method successfully calculates the amplification factor for H=2.44. Displayed is the calculated suction distribution for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L}=3.01\cdot 10^6$ and $c_l=0.45$.



for H = 2.40. Displayed is the calculated suction distribution for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$ and $c_l = 0.45$.

3.2.2 Broers

Broers [2004] modified the version of Ferreira, added new features and made it more user friendly. He added the functionality to calculate suction over a part of the airfoil. He also implemented an improved command to quickly calculate the basic suction distribution, according to Equation (2.35), using the potential or viscous flow pressure distribution and treating Re_{θ} as a constant or a variable, depending on the settings.

A main feature is the ability to calculate a constant shape factor boundary layer development (laminar or turbulent) for a certain angle of attack or lift coefficient. This is an iterative process where the program searches for an optimal suction distribution within limits specified by the user. Also it is possible to calculate automatically the suction distributions for a range of angles of attack or lift coefficients.

3.2.3 Bongers

Bongers [2006] made the most recent version of XFOIL-suction. He implemented the Van Ingen method in a completely different way, but using exactly the same theory. This now called Improved Van Ingen method, is able to calculate the amplification factor correctly. In Figure 3.5 the same situation is present as in Figure 3.4, but now the Improved Van Ingen method is used. Clearly can be seen that the amplification factor decreases, whereas the 'normal' Van Ingen method keeps it constant.



factor for H = 2.40. Displayed is the calculated suction distribution for the BW 10–144 at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$ and $c_l = 0.45$.

Bongers also improved the structure of the program, giving the user the ability to select the transition method (Drela, Van Ingen, Improved Van Ingen) and its convergence method. He also changed the colors in the graphical user interface, making the printouts more readable.

Throughout this thesis the XFOIL-suction version of Bongers and the Improved Van Ingen method is used because it calculates the most accurate results. From now on, this version is implied when the text refers to XFOIL, unless mentioned otherwise.

3.3 Calibrating XFOIL

XFOIL has various parameters to model the flow. To correctly calculate the airfoil's properties, some parameters have to be determined. These parameters are generally dependent on Reynolds number and flight conditions. The parameters that have to be determined are the critical amplification factor N_{crit} and the shear lag constant SCC.

The critical amplification factor N_{crit} is the factor which determines the point where the boundary layer transits from a laminar one into turbulent one in the e^n method of Van Ingen. It depends on the disturbance level in which the airfoil operates, i.e. the amount of turbulence of the ambient flow. In practice, N_{crit} influences the drag and the width of the low-drag bucket.

The shear lag factor SCC determines when the flow separates from the airfoil in the

| Re | N_{crit} | SCC | Δc_d |
|----------------|------------|-----|----------------------|
| $3 \cdot 10^6$ | 10.0 | 4.0 | $5.17\cdot 10^{-4}$ |
| $6\cdot 10^6$ | 10.0 | 4.0 | $5.17\cdot 10^{-4}$ |
| $9 \cdot 10^6$ | 10.0 | 4.0 | $5.17 \cdot 10^{-4}$ |

Table 3.1: Calibration factors for XFOIL

calculations. It determines when the non-linear part of the lift curve starts and how it develops. In practice, SCC determines the maximum lift coefficient.

To determine N_{crit} and SCC, wind tunnel results of the NACA 63_2 -415 airfoil are used (Abbott & Von Doenhoff [1959]). This airfoil is chosen because the original wing of the Eaglet is equipped with it from root to tip. The polars of the NACA 63_2 -415 airfoil, for Reynolds numbers of $3 \cdot 10^6$, $6 \cdot 10^6$ and $9 \cdot 10^6$, can be found in Figure C.1 and Figure C.2 in Appendix C.

The results of the calibration are given in Table 3.1 and the comparison with Abbott & Von Doenhoff can be found in Figures D.1 to D.6 in Appendix D. Even after setting the right parameters, XFOIL is too optimistic in determining the drag. To compensate this, also an additional drag coefficient $\Delta c_d = 5.17 \cdot 10^{-4}$ is determined to compensate the final results as will be explained later on. This correction is applied in Figures D.1 to D.6 and it can be seen that the corrected XFOIL polars are in close agreement with the wind tunnel data.

3.4 Creating a Batch Generator for Normal Calculations in XFOIL

Many analyses have to be done with XFOIL. To get polars with good accuracy, a small step in angle of attack is chosen². The problem with XFOIL is however, that when it cannot converge for four subsequent angles of attack or lift coefficients, it will terminate its calculations and therefore not finish the polar. Another problem in XFOIL is that it sometimes has the tendency to freeze and with it to loose all the data it calculated before. This implied a lot of manual input and a lot of extra time needed. Therefore a batch generator is created in Matlab. It sets the required conditions, and instead of making an α or c_l -sweep, it calculates the points one by one and therefore calculates as many points as possible. However, sometimes it still crashes. In that case the problem causing α or c_l value has to be excluded from the calculations. A flow chart of the batch generator is shown in Figure 3.6.

3.5 Creating a Batch Generator for Suction Calculations in XFOIL

XFOIL has three ways of calculating polars, named Type 1, 2 and 3. Type 1 calculations simulate the condition of a wind tunnel; a given airfoil at a constant velocity undergoes

²For all calculations $\Delta \alpha = 0.25^{\circ}$ is used.



Figure 3.6: Flow diagram of the XFOIL batch program

an angle of attack change and thus changing its lift. Type 2 calculations correspond to a level flight situation; a given airfoil at a constant lift undergoes an angle of attack change resulting in a change in velocity. Finally Type 3 calculations correspond to a 'rubber chord'; the velocity and lift are kept constant and the airfoil undergoes an angle of attack change, resulting in a variable chord.

The Type 2 method is chosen to simulate flight conditions. This means that the angle of attack is varied while keeping the lift (i.e. weight of the aircraft) constant, implying that the velocity is variable. To keep the lift constant, $\text{Re}\sqrt{C_L}$ has to be kept constant, as derived below:

$$L = C_L \frac{1}{2} \rho V^2 S \qquad \text{Re} = \frac{\rho V c}{\mu}$$
$$L = C_L \frac{1}{2} \rho V^2 S = \text{constant}$$
$$C_L \frac{1}{2} \rho \left(\frac{\text{Re}\mu}{\rho c}\right)^2 S = \text{constant}$$
$$\text{Re}\sqrt{C_L} = \text{constant}$$

In XFOIL, the suction distribution is made dimensionless by dividing the vertical velocity v_0 by the free stream velocity U_{∞} . In a practical situation however, it is shown in Broers [2004] that the vertical velocity v_0 is constant. Because in XFOIL v_0 can not be defined explicitly only its dimensionless value $\frac{v_0}{U_{\infty}}$, for every lift coefficient a new suction distribution $\frac{v_0}{U_{\infty}}$ has to be specified. This is a lot of work for all the situations that have to be analyzed, especially when a small step in angle of attack is chosen. To prevent this, another batch generator is written in Matlab. Essentially it reads in a previously calculated polar without suction, and calculates for every c_l the corresponding suction distribution by scaling the (tailored) suction distribution $\frac{v_0}{U_{\infty}}$ in such a way that v_0 is constant. Thereafter it calculates a new polar, with suction and new values of c_l . This new polar is read in and the same progress starts over until sufficient accuracy is achieved. From experiments it followed that three iteration loops were sufficient to get good accuracy for a small step size in angle of attack. The flow diagram of the batch generator can be seen in Figure 3.7.

3.6 XFLR

XFLR is a program which uses Drela's XFOIL code to determine the sectional properties of airfoils, and extends it to three dimensions using Lifting Line Theory, Vortex Lattice Methods or a 3D Panel Method (Deperrois [2009]). For calculations of the wing, lifting line theory is chosen because its implementation in XFLR is a non-linear one which uses viscous data and is more reliable given its limitations compared to the other methods. The downside of this method is that it does not take into account the thickness of the wing. XFLR uses a non-linear method described in Sivells & Neely [1947]. Its limitations



Figure 3.7: Flow diagram of the XFOIL batch program for suction



Figure 3.8: A lifting line with vortex sheet, used in lifting line theory (Anderson Jr. [2001])

are that the wing should not have a too low aspect ratio and a large amount of sweep. The wing of the Eaglet is well within these limits.

In lifting line theory the wing is modeled by a lifting line, formed by horseshoe vortices of variable strengths. The horseshoe vortex consists of a bound vortex, which is fixed at the wing's position, and two trailing vortices which extend to infinity in the wake. The strength of an individual horseshoe vortex is constant due to Helmholtz's first theorem³. The trailing vortices exist because a vortex cannot terminate in the fluid due to Helmholtz's second theorem⁴. The superposition of these horseshoe vortices will yield a lifting line with variable strength in spanwise direction and a vortex sheet consisting of free trailing vortices, as can be seen in Figure 3.8.

The local lift is expressed as a function of the local circulation strength:

$$l(y) = \rho_{\infty} V_{\infty} \Gamma(y)$$

The trailing vortices induce a downward velocity which will reduce the effective angle of attack and tilt the local lift vector backwards, introducing a force called the induced drag, see Figure 3.9.

In contrast to linear lifting line theory, where the lift is assumed a linear function of angle of attack, XFLR uses the method developed by Sivells & Neely [1947]. It is a non-linear method which uses experimental or calculated sectional lift data instead of assuming a linear relation. XFLR is able to calculate the sectional data because Drela's XFOIL code is implemented in the program. However, because a modified version of XFOIL is used in this thesis, the sectional data is calculated with the Bongers version of XFOIL and imported into XFLR.

For calculating the lift and drag there are two general methods, the near field and the far field method. Near field calculations consist of directly integrating the pressure over the surface. This will yield inaccurate results because the integration is very sensitive to

³Helmholtz's first theorem: The strength of a vortex filament is constant along its length.

 $^{^{4}}$ Helmholtz's second theorem: A vortex filament cannot end in a fluid; it must extend to the boundaries of the fluid or form a closed path.



Figure 3.9: The effect of downwash due to the downward velocity induced by the trailing vortices (Anderson Jr. [2001])

errors. XFLR uses the far field method, which is based on the balance of momentum in a far field (Trefftz) plane. This method gives an accurate result.

Chapter 4

Characteristics of the Original Wing

To determine the improvements of the wing that has yet to be designed, the characteristics of the original wing are needed. Only flight test data of the complete aircraft is available, but not of the wing separately. In order to investigate the improvements, the wing has to be isolated from the aircraft. This is done by calculating the original wing properties and subtracting this from the flight test data, resulting in the lift and drag of the aircraft minus the wing. This is a rather rough approach, however the best way with the tools and data available. Moreover, the wing layout is not changed dramatically, making it less sensible to interference effects. When the new wing is designed it can replace the old wing data, giving the approximate results of the aircraft with the new wing. In this way the improvements of the entire aircraft can be analyzed.

The wing is calculated with the method of Diederich [1952], using the handbook of Torenbeek [1982] as a guideline. The wing is also calculated with XFLR to verify the results. Some key properties of the original wing are summarized in Table 4.1, for more details see Appendix A.

| Area: | 9.85 $[m^2]$ |
|--------------------------|-----------------|
| Aspect ratio: | 7.02 [-] |
| Root chord: | 1.530 [m] |
| Tip chord: | 0.840 [m] |
| Taper ratio: | 0.549 [-] |
| Root incidence: | 3.00 [°] |
| Tip incidence: | 0.50 [°] |
| Sweep angle at $0.25c$: | 0.00 [°] |
| Mean geometric chord: | 1.185 [m] |
| Root airfoil: | NACA 63_2-415 |
| Tip airfoil: | NACA 63_2-415 |

Table 4.1: Properties of the original wing (Anonymous [2009])

4.1 Flight Test Data

The characteristics of the Eaglet are available in the form of fight test data. This data is obtained by performing measurements of the necessary parameters at different flight conditions (Melkert [2000]). All the data is corrected to ISA conditions and the lift and drag is calculated. The lift curve of the Eaglet can be found in Figure 4.1 and the drag polar in Figure 4.2. In general the linear part of the lift curve of an aircraft can be approximated by the following expression:

$$C_L = C_{L_\alpha} \alpha + C_{L_0}$$

Fitting a linear line through the flight test data, Figure 4.1, yields the expression for the lift curve of the Eaglet:

$$C_L = 3.7377\alpha + 0.4434$$

In the same way the drag polar of an aircraft can be approximated by:

$$C_D = \frac{kC_L^2}{\pi A} + C_{D_0}$$

Fitting a linear line through the C_D versus C_L^2 will yield the expression of the drag polar of the Eaglet:

$$C_D = 0.0577 C_L^2 + 0.0457$$

4.2 Lift and Drag Calculations

Torenbeek's handbook consists of a compilation of methods based on theoretical and empirical results, often simplified for certain conditions. It therefore dictates some restrictions on geometry and flight conditions. These restrictions are summarized below:

- 1. flight speeds are subcritical
- 2. angles of attack are relatively small
- 3. wing aspect ratios exceed $\frac{4}{\cos \Lambda_{\frac{c}{2}}}$ and wing sweep angles are less than 35°
- 4. only power-off conditions are considered
- 5. effects of aero-elasticity are ignored
- 6. ground effects are not considered



Figure 4.1: The lift curve of the Eaglet (power off conditions)



Figure 4.2: The drag polar of the Eaglet (power off conditions)

The normal operating conditions of the Eaglet are well within subcritical conditions. For example the manoeuvre and never exceed speed yield Mach-numbers of 0.20 and 0.27 respectively at sea-level conditions. Also the angle of attack during cruising conditions is small. The aspect ratio of the wing is 7.02, which is larger than $\frac{4}{\cos \Lambda_{\frac{2}{4}}} = 4$ and the wing sweep equals to zero, see also Table 4.1. The last three conditions are assumed during the calculations; only power-off conditions are considered, and aero-elastic and ground effects are neglected.

4.2.1 Lifting Properties of Airfoil Sections

The wing of the Eaglet has a NACA 63_2 -415 airfoil from root to tip. To determine the zero-lift angle and the lift-curve slope of this profile, Abbott & Von Doenhoff [1959] is used, see also Figure C.1 in Appendix C. Clearly can be seen that the zero-lift angle α_{l_0} is equal to -3° , independent of Reynolds number. The lift-curve slope can be determined by taking the difference in c_l over an α -range within the linear part of Figure C.1, see Equation (4.1).

$$c_{l_{\alpha}} = \frac{\mathrm{d}c_{l}}{\mathrm{d}\alpha} \approx \frac{\Delta c_{l}}{\Delta \alpha} = \frac{1.4 - (-0.6)}{10 - (-8)} \frac{1}{\frac{\pi}{180}} = 6.366 \ \mathrm{rad}^{-1}$$
(4.1)

4.2.2 Wing Lift

The lift-curve slope of the complete wing can be determined by the following equation (Anderson [1936]):

$$C_{L_{w_{\alpha}}} = f \frac{c_{l_{\alpha}}}{E + \frac{c_{l_{\alpha}}}{\pi A}} \tag{4.2}$$

Besides $c_{l_{\alpha}}$, also the variables f, E and A are needed. Variable f, the correction factor for wing taper, is shown in Figure 4.3. Using the data of the Eaglet, A = 7.02 and $\lambda = \frac{c_t}{c_{\alpha}} = 0.549$, will yield an f of 0.999.

The Jone's edge velocity factor E can be approximated by:

$$E = 1 + \frac{2\lambda}{A(1+\lambda)}$$

Using the Eaglet data will yield E = 1.101. The wing lift-curve slope, Equation (4.2) can now be calculated and is equal to $C_{L_{w_{\alpha}}} = 4.576 \text{ rad}^{-1}$.

4.2.3 Lift Distribution

The spanwise lift distribution can be divided into an additional and a basic lift distribution:

$$c_l = c_{l_a} + c_{l_b}$$



Figure 4.3: Correction factor for wing taper (Abbott & Von Doenhoff [1959])

This can be rewritten using Anderson's lift functions L_a and L_b , in the following way:

$$c_l \frac{c}{c_g} = L_a C_L + \frac{\epsilon_t c_{l_\alpha}}{E} L_b \tag{4.3}$$

Where in Equation (4.3), L_a and L_b are respectively

$$L_a = \frac{c_{l_a}}{C_L} \frac{c}{c_q} \tag{4.4}$$

and

$$L_b = c_{l_b} \frac{c}{c_g} \frac{E}{\epsilon_t c_{l_\alpha}}$$

According to Diederich [1952], Equation (4.4) can be rewritten in terms of coefficients C_1 , C_2 and C_3 , see Figure 4.4:

$$L_a = C_1 \frac{c}{c_g} + (C_2 + C_3) \frac{4}{\pi} \sqrt{1 - \eta^2}, \qquad (4.5)$$

where η is the non-dimensional spanwise coordinate, $\eta = \frac{2y}{b}$. Diederich's method is a semiempirical approach and is valid for wings with arbitrary planform and lift distribution, given that the quarter-chord line of the wing is approximately straight.

Planform parameter F in Figure 4.4 is defined in the following way:

$$F = \frac{2\pi A}{c_{l_{\alpha}} \cos \Lambda_{\frac{c}{4}}}$$

Using the data of the Eaglet gives F = 6.929, yielding:



Figure 4.4: Coefficients in Diederich's method (Diederich [1952])

$$C_1 = 0.34$$

 $C_2 = 0.37$
 $C_3 = 0.28$

Finally, the chord c in Equation (4.5) is given by:

$$c = c_r + (c_t - c_r)\eta$$

In a similar way L_b can be rewritten:

$$L_b = \beta E L_a C_4 \cos \Lambda_\beta \left(\frac{\epsilon}{\epsilon_t} + \alpha_{0_1}\right) \tag{4.6}$$

where:

$$\alpha_{0_1} = -\int_0^1 \frac{\epsilon}{\epsilon_t} L_a \,\mathrm{d}\eta = -0.430,$$

$$\epsilon = \epsilon_r + (\epsilon_t - \epsilon_r)\eta$$

and $C_4 = 0.46$, see Figure 4.5.

In Equation (4.6), Λ_{β} is the corrected sweep angle, which equals to zero due to the lack of sweep of the Eaglet's wing. The Prandtl-Glauert correction β is calculated for one



Figure 4.5: Coefficient C_4 in Diederich's method

reference Mach-number, the cruise Mach-number at zero altitude ISA. The cruise speed can be found in Poels [1998] and equals to $V_c = 55.6 \ \frac{m^1}{s}$. This yields a Mach-number of 0.163 and a Prandtl-Glauert correction factor of 0.987 $\left(\beta = \sqrt{1-M^2}\right)$.

In Figure 4.6, Anderson's lift functions L_a and L_b are plotted against the span. With L_a and L_b known, c_{l_a} , c_{l_b} and c_l can be calculated:

$$c_{l_a} = L_a C_L \left(\frac{c}{c_g}\right)^{-1}$$
$$c_{l_b} = L_b \frac{\epsilon_t c_{l_\alpha}}{E} \left(\frac{c}{c_g}\right)^{-1}$$
$$c_l = \left(\frac{c}{c_g}\right)^{-1} \left(C_L L_a + \frac{\epsilon_t c_{l_\alpha}}{E} L_b\right)^{-1}$$

The results for cruise conditions are shown in Figure 4.7. More lift distributions can be found in Appendix E.

4.2.4 Lift-curve of the Wing

The lift-curve of the complete wing can be calculated with:

¹At cruise, $C_L = 0.45$ at $\frac{W}{S} = 846.26 \frac{N}{m^2}$







Figure 4.7: Lift distribution for cruise conditions ($C_L = 0.45$)



Figure 4.8: Calculated lift-curve of the original wing of the Eaglet

$$C_{L_w} = C_{L_{w\alpha}} \left\{ \alpha_r - (\alpha_{l_0})_r - \alpha_{0_1} \epsilon_t \right\}$$

where:

$$\alpha_r = \alpha + i_r$$

With all variables known, the lift-curve of the total wing can be plotted, as can be seen in Figure 4.8.

4.2.5 Induced Drag

The induced drag of an untwisted wing can be calculated by:

$$C_{D_i} = (1+\delta)\frac{C_L^2}{\pi A}$$

Because the wing of the Eaglet is twisted, this result has to be corrected later on. Torenbeek [1982] gives two methods to calculate the δ factor, the method of Garner [1968] and the method of Anderson [1936]. The method of Anderson is chosen, because its result is closer to the one displayed in Figure F.1 in Torenbeek [1982].

$$\delta = \left\{ 0.0015 + 0.016(\lambda - 0.4)^2 \right\} (\beta A - 4.5) = 0.0045$$



Figure 4.9: Induced-drag factor v (Abbott & Von Doenhoff [1959])

To compensate for wing twist, Anderson suggests the following expression:

$$\Delta_{\epsilon} C_{D_i} = C_L \left(\frac{\epsilon_t c_{l_{\alpha}}}{E}\right) v + \left(\frac{\epsilon_t c_{l_{\alpha}}}{E}\right)^2 w$$

where v and w are the induced-drag factors, which can be found in Figure 4.9 and Figure 4.10. Factor v equals to 0.0008 and w to 0.0038.



Figure 4.10: Induced-drag factor w (Abbott & Von Doenhoff [1959])



Figure 4.11: Profile drag coefficient of the total wing

4.2.6 Profile Drag

To determine the profile drag of the total wing, the sectional profile drag is integrated over the wing span, using Equation (4.7).

$$C_{D_p} = \frac{2}{S} \int_0^{\frac{b}{2}} c_{d_p} c \,\mathrm{d}y \tag{4.7}$$

Because the wing is tapered (variable Re number) and c_{d_p} is a function of c_l , and in turn c_l is different over the span for a given C_L , many Re number and c_l combinations are necessary. Because of the way the new wing will be designed and analyzed it is useful to calculate the values of c_{d_p} with XFOIL.

In XFOIL, for various flight conditions ($\text{Re}\sqrt{C_L} = \text{constant}$) the polars are calculated and results in between are linearly interpolated since the data is dense enough to yield only marginal errors.

The profile drag of the wing is calculated with Equation (4.7) and is shown in Figure 4.11.

4.2.7 Total Drag

Now the components of the drag are calculated, the total drag can be calculated as shown in Equation (4.8).

$$C_D = C_{D_i} + \Delta_{\epsilon} C_{D_i} + C_{D_p} \tag{4.8}$$



Figure 4.12: The drag breakdown of the complete wing

The components C_{D_i} and $\Delta_{\epsilon} C_{D_i}$ account for the induced drag, and the profile drag C_{D_p} accounts for skin friction and pressure drag.

The drag breakdown can be seen in Figure 4.12. Figure 4.13 shows the relative importance of C_{D_i} and C_{D_p} . At cruise conditions ($C_L = 0.45$), 64% of the wing drag is due to induced drag and 36% is due to profile drag.

4.3 Wing calculations with XFLR

To validate the lift and drag characteristics obtained in the previous section, the wing is also calculated in XFLR and compared with the method of Torenbeek. The results are shown in Figure F.1 to Figure F.4 in Appendix F. As can be seen in Figure F.1, the method described by Torenbeek uses a linear approximation for C_L whereas XFLR uses a non-linear method. However, this yield only small errors in the drag curves as can be seen in the subsequent figures. The drag prediction of the method described by Torenbeek and XFLR show the same results, see Figure F.2 to Figure F.4. XFLR will be used to calculate the lift and the drag of the new wing. Its results will be presented in Chapter 7.

4.4 Results

Now both the flight test data and the wing data are calculated, they can be compared with each other. The wing is simply subtracted of the total aircraft to get the properties



Figure 4.13: The relative drag breakdown of C_{D_i} and C_{D_p} of the wing

of the aircraft minus the wing. This is a rather rough estimate of the aircraft minus wing because interaction between the two is significant. However, when the new wing is calculated, it will be added to the same 'aircraft minus wing', and the interactions will cancel out for a large portion.

Figure 4.14 and Figure 4.15 show respectively the lift and the drag of the aircraft, wing and aircraft minus the wing. It can be seen that the lift of the aircraft is a bit decreased due to the horizontal tail-plane, therefore the lift curve of the total aircraft is a bit less than the one of the wing. In Figure 4.15 can be seen that the aircraft minus wing has a rather constant drag over the flight regime.

Figure 4.16 is very interesting for this thesis, it shows the profile drag of the wing with respect to the total aircraft. It gives insight how much effect a reduction in profile drag due to boundary layer suction would have. It is disappointing to notice that the profile drag of the wing contributes only for a very small part to the total drag of the aircraft. This is due to a far from optimal design of the fuselage, which generates a lot of the drag. The profile drag of the wing as a percentage of the total aircraft drag is shown in Figure 4.17. This figure very clearly shows that the profile drag of the wing is about 8 to 10% of the total aircraft drag. Meaning for cruise conditions ($C_L = 0.45$), that if the profile drag of the wing could be reduced by 50%, the total aircraft drag would only be reduced by about 4.5%. Nontheless it is a very interesting case to examine, starting with the design of a new airfoil as will be explained in Chapter 5.



Figure 4.14: The lift-curve of the individual parts of the Eaglet



Figure 4.15: The drag of the individual parts of the Eaglet



Figure 4.16: The profile drag of the wing with respect to the total aircraft drag



Figure 4.17: The relative wing profile drag with respect to the total drag of the aircraft
Chapter 5

The New Airfoil

5.1 The Requirements

Before designing the new airfoil, the requirements have to be determined. The goal of the new wing is cruising with a low profile drag by applying boundary layer suction on the upper side and having large regions of natural laminar flow at the lower side. A relative large area of natural laminar flow on the upper side of the airfoil is also beneficial. The more natural laminar flow exists (passive), the less suction has to be applied (active), thus saving energy.

For practical reasons, it is necessary that suction distribution over the chord has a relative simple shape (constant, triangular) to simplify the implementation of the system. To install the suction equipment in the wing, a certain amount of height is needed, especially at the rear part of the airfoil, and therefore a too thin airfoil is unwanted.

The Eaglet is a general aviation aircraft, which in general is flown by private pilots whom have had less training and flying experience than commercial pilots. Therefore, the aircraft's handling characteristics have to be smooth and forgiving. Gradual stalling characteristics are therefore essential for the new wing. Finally, the new wing has to function as well when the suction is turned off or is malfunctioning, and preferably with better performance than the original one.

5.2 The BW 10–144 Airfoil

When designing a new airfoil, it is necessary to find a good starting point. An airfoil has to be found which in general has the properties needed, which later can be altered and tailored to meet the design requirements. In the case for an airfoil for boundary layer suction, the catalogs Althaus [1972] and Althaus [1996] are good starting points.

For the upper side of the airfoil it is useful to have a flat-plate boundary layer for a large portion of the chord. A flat-plate boundary layer has a constant pressure distribution and therefore no adverse pressure gradient, promoting stability of the laminar boundary layer for a relative large percentage of the chord. The Wortmann FX S 03–182 forms a good base for the upper side of the new airfoil. The profile and potential flow pressure distributions of the FX S 03–182 are shown in Figure G.1 and Figure G.2 in Appendix G. The airfoil has a favorable pressure gradient up to 50% chord, up to 4 degrees angle of attack, promoting laminar flow.

Boundary layer suction will not be installed on the lower side of the wing. Therefore it is favorable to have a large region of natural laminar flow. The Wortmann FX 38–153, found in Althaus [1972], is a good candidate for the lower side of the airfoil. Its shape and potential flow pressure distributions can be found in Figure G.3 and Figure G.4 in Appendix G. It has a favorable pressure gradient up to 70% chord, promissing a large area of laminar flow.

As described before, the base of the new airfoil is formed by the Wortmann FX S 03–182 at the upper side and the Wortmann FX 38–153 at the lower side. After numerous manipulations the BW 10–144 airfoil arose. Its shape and potential flow pressure distribution can be found in Figure G.5 and Figure G.6 in Appendix G. The airfoil has a 14.4% thickness, with a large region of laminar flow on the lower side and an upper side with laminar flow up to 55% chord.

Comparing the BW 10–144 to the NACA 63_2 –415 airfoil at a representative Reynolds number, see Figure 5.1, it can be seen that the BW 10–144 has lower or equal drag for lift coefficients between 0.2 and 1.2. However, its maximum lift coefficient is about 0.03 lower. The moment coefficient is lower than the one of the NACA 63_2 –415 airfoil. At the upper side, the BW 10–144 has laminar flow beyond 50% chord up to a lift coefficient of 0.7 and the lower side has laminar flow from 70 up to $75^{1}\%$ chord at lift coefficients higher than 0.3.

5.3 The Suction Distribution

As will be explained later, the wing will be equipped with a 22% chord flap and aileron. Because in practice it is not feasible to implement boundary layer suction on the flap or aileron, suction is only applied on the wing. In practice, the transition to a flap or aileron will not disturb the flow sufficiently to let the boundary layer transit to a turbulent one. To control this, the boundary layer is deliberately tripped to a turbulent one at 75% chord at both the upper and lower side of the airfoil. This tripping of the boundary layer can be done with the use of zig-zag tape. The transition point is chosen at 75% chord to give the boundary layer time to transit to a turbulent one before it reaches the flap or aileron. This 'fresh' turbulent boundary layer will be able to cope with relative high pressure gradients, making the flap more effective. The polars of the BW 10–144 airfoil with free and forced transition, at a representative Reynolds number, can be found in Figure 5.2.

Amongst other information, Figure 5.2 shows the free and fixed transition point on the upper and lower side of the airfoil. As can be seen, on the upper side of the airfoil the transition point is at 50-55% chord, up to a lift coefficient of about 0.7. After this lift

¹Forced transition at 75% chord, see the next section.



Figure 5.1: Polars of the NACA 632-415 and BW 10-144 airfoil (MGC)



Figure 5.2: Polars of the BW 10-144 airfoil with free (red) and forced (blue) transition



Figure 5.3: Non-scaled suction distribution for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$

coefficient the transition point gradually moves forward. To get laminar flow up to and in many cases beyond this lift coefficient, with some margin to prevent transition, suction has been applied from 40% chord onwards to 75% chord. The restriction at 75% chord is due to the flap or aileron, as explained above.

In XFOIL the optimal suction distribution has been calculated for various lift coefficients, as can be seen in Figure 5.3. They have the same general shape, apart from their magnitude. Broers [2004] showed that for flight situations (Type 2 calculation is XFOIL), the suction distributions can be scaled by a factor of $c_l^{-\frac{1}{2}}$. In doing so, see Figure 5.4, the suction distributions almost become equal, meaning that the v_0 distribution is the same at all the c_l values. Using this fact, the same suction distribution can be used for every value of c_l , making practical implementation less complicated.

Figure 5.3 shows suction distributions of a rather smooth shape. In practice however it is very difficult to capture all the detail of these distributions, a more general shape has to be applied. Figure 5.5 shows this tailored suction distribution in its unscaled, dimensionless form. Its maximal magnitude is $-1 \cdot 10^{-5}$, which can be scaled in XFOIL to the desired value. The tailored suction distribution consists of a linear shape from 40 up to 55% chord, followed by a constant part until 75% chord. To compare it with the calculated distributions, it is also plotted in Figure 5.4.

Figure 5.6 shows a suction distribution at cruise conditions as calculated by XFOIL. The suction is applied from 40 to 75% chord, as explained above. XFOIL has calculated a constant shape factor boundary layer of H = 2.6, which represents a flat plate boundary layer. In Figure 5.7, the calculated suction distribution is replaced by the tailored one. As can be seen, its shape factor is still rather constant and after an initial increase due to



Figure 5.4: Scaled suction distribution for the BW 10–144 airfoil at ${\rm Re}\sqrt{C_L}=3.01\cdot 10^6$



Figure 5.5: Tailored suction distribution for the BW 10-144 airfoil



more initial suction, the amplification factor decreases, staying far from the critical value of 10. Also, in both cases it can be seen that the boundary layer is forced to trip at 75% chord, where the shape factor decreases to H = 1.4, which is typical for a turbulent flat plate boundary layer.



Figure 5.7: Tailored suction distribution, BW 10-144 airfoil in cruise conditions

Chapter 6

The New Wing

To keep the flight characteristics of the Eaglet similar, the general shape of the wing is unchanged. The taper ratio of $\lambda = 0.549$ gives good induced drag properties (Hoerner [1965]), while keeping the lines of the wing rather straight. This is important to keep the installation necessary for suction simple, without too much variation in spanwise direction.

For stability reasons, also the dihedral has kept its original value of $\Gamma = 5^{\circ}$. The wing twist ϵ is removed however. The reason for implementing wing twist is to let the stall of the wing begin at the root. This ensures roll control of the aircraft at initial stall. However, the stalling characteristics of the BW 10–144 airfoil are very gradual, making the stall of the wing also gradual. Moreover, the lift distribution over the wing, for example see Figure E.6 in Appendix E shows that the local lift coefficients near the tips are lower than more inboard, making it stall first at the more inboard stations. In addition, due to crossflow caused by the fuselage, the angle of attack at the wing root is higher than more outboard, causing separation to start at the wing root first.

Another reason for removing the twist is to further simplify the implementation of suction. Without twist, the spanwise stations have nearly the same pressure distribution and therefore the amount of suction needed only depends on the local Reynolds number, not on the individual pressure distributions locally.

The general shape of the new wing is shown in Figure 6.1 and the details are given in Table 6.1.

6.1 Method of Determining the Amount of Suction

The amount of suction needed only depends on the chord length (Reynolds number). In the ideal case, every spanwise station has its own suction distribution, in practice this is too complicated however. For a limited amount of stations, only one optimum suction distribution can be implemented. For layout reasons, the wing is subdivided in four

| Fuselage | | Station 1 | Station 2 | Station 3 | Station 4 |
|----------|---|-----------|-----------|-----------|-----------|
| | Γ | FI | ар | Aile | eron |
| | | | | | |

Figure 6.1: Schematic representation of the right wing

| Area: | 9.85 [m ²] |
|--------------------------|------------------------|
| Aspect ratio: | 7.02 [-] |
| Root chord: | 1.530 [m] |
| Tip chord: | 0.840 [m] |
| Taper ratio: | $0.549 \ [-]$ |
| Root incidence: | $2.00 [^{\circ}]$ |
| Tip incidence: | 2.00 [°] |
| Sweep angle at $0.25c$: | 0.00 [°] |
| Mean geometric chord: | 1.185 [m] |
| Root airfoil: | BW 10–144 |
| Tip airfoil: | BW 10–144 |

Table 6.1: Properties of the new wing



Figure 6.2: c_l factor criterion for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$

stations, and its ideal amount of suction is determined. These four stations consist of two 'flap' stations and two 'aileron' stations. Later on a decision will be made for the final layout. Figure 6.1 shows a schematic drawing of the wing with its stations.

No suction will be applied in the inboard section of the wing (between the fuselage and the flaps). This is because the inner wing is partly turbulent due to a turbulent wedge of about 10° starting at the root's leading edge. Besides, the wing is equipped with an anti-skid layer to prevent slipping while embarking the airplane, see Figure B.1 in Appendix B. This layer has an enormous roughness and the boundary layer will be triggered to turbulent immediately. Finally the propeller is feeding the inner wing with a partial turbulent flow, originating from the propeller blade's wake.

As explained in Section 5.3, a tailored suction distribution will be used and scaled for the right conditions. To determine the maximum lift coefficient and the amount of suction needed at a certain Reynolds number, for which the boundary layers stays laminar until 75% chord, two criteria are used. To be certain that the boundary layer is not close to transition before it enters the suction region, a minimum N factor of 8 at 40% chord is used as a c_l criterion. This is shown in Figure 6.2. It can be seen that the airfoil without suction has an N factor of 8 at 40% chord at $c_l = 0.73$. This gives some margin and prevents premature transition.

For determining the amount of suction and therefore the scaling of the tailored suction distribution, the following criterion is used. The amplification factor must remain around



Figure 6.3: Scaling criterion for the BW 10–144 airfoil at $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$

8 to have some margin before transition, and at the end of the suction region (75% chord) the amplification factor must not be much higher than 8. This criterion is clarified in Figure 6.3.

The effect of boundary layer suction can be clearly seen in the difference in c_d between Figure 6.2 ($c_d = 0.00473$) and Figure 6.3 ($c_d = 0.00267$); with suction the drag is only 56% of the drag without suction. Also the change in angle of attack at the same lift coefficient can be seen, this is why the batch generator of Section 3.5 needs to make some iterations.

6.2 The Spanwise Suction Distribution

Starting with the suction equation (2.35) derived in Chapter 2, the spanwise suction distribution can be calculated.

$$\frac{v_0}{U_{\infty}} = \frac{1}{0.2205} \frac{1}{\frac{U}{U_{\infty}}} \frac{\mathrm{d}\left(\frac{U}{U_{\infty}}\right)}{\mathrm{d}\left(\frac{x}{c}\right)} \frac{\mathrm{Re}_{\theta}}{\mathrm{Re}_c}$$
(6.1)

Using the by Blasius calculated momentum loss thickness for a flat plate (White [2006]):

$$\theta = 0.664 \frac{x}{\sqrt{\frac{Ux}{\nu}}},$$

and noticing that the momentum loss thickness at the start of the suction area must be proportional to this, the Reynolds number based on the momentum loss thickness can be rewritten:

$$\operatorname{Re}_{\theta} = \frac{U\theta}{\nu} \propto \frac{U}{\nu} \frac{x}{\sqrt{\frac{Ux}{\nu}}}$$
$$\operatorname{Re}_{\theta} \propto \sqrt{\operatorname{Re}_{x}}$$

Substituting this result into Equation (6.1) leads to:

$$\frac{v_0}{U_{\infty}} \propto \frac{1}{\frac{U}{U_{\infty}}} \frac{d\left(\frac{U}{U_{\infty}}\right)}{d\left(\frac{x}{c}\right)} \frac{\sqrt{\text{Re}_x}}{\text{Re}_c}$$
$$\frac{v_0}{U_{\infty}} \propto \sqrt{\frac{\frac{x}{c}}{\frac{U}{U_{\infty}}}} \frac{d\left(\frac{U}{U_{\infty}}\right)}{d\left(\frac{x}{c}\right)} \frac{1}{\sqrt{\text{Re}_c}}$$

Inserting dimensionless variables $x' = \frac{x}{c}$ and $U' = \frac{U}{U_{\infty}}$ gives the final result:

$$\frac{v_0}{U_{\infty}} \propto \sqrt{\frac{x'}{U'}} \frac{\mathrm{d}U'}{\mathrm{d}x'} \frac{1}{\sqrt{\mathrm{Re}_c}}$$

Because the pressure distributions over the span of the wing are the same, due to the lack of twist, also the dimensionless velocity U' is the same at a certain x' position. Therefore the suction velocity is proportional to the inverse of the square root of the Reynolds number:

$$\frac{v_0}{U_{\infty}} \propto \frac{1}{\sqrt{\text{Re}_c}} \tag{6.2}$$

This observation is verified by calculating the required suction distributions on several spanwise positions. The results are plotted in Figure 6.4 and also a linear line is fitted through this data. Clearly the resemblance between Equation (6.2) and the data plotted in Figure 6.4 can be seen.

6.3 Suction Distributions at Various Spanwise Stations

For the root, the spanwise stations and the tip, the suction distributions have been determined. The criteria defined in Section 6.1 are used. Figure 6.5 shows the polars of



Figure 6.4: The spanwise suction distribution

the BW 10–144 airfoil with and without suction and the NACA 63_2 –415 airfoil without suction. The Re $\sqrt{C_L}$ corresponds to a flight situation, based on the mean geometric chord.

Clearly shown is the huge amount of profile drag reduction due to the suction. Comparing the BW 10–144 airfoil with and without suction shows a large reduction in profile drag. The airfoil with suction has only 63.5% of the profile drag of the one without suction at a cruising lift coefficient of 0.45. Comparing it to the original airfoil, the NACA 63_2 –415, this is only 52.4%, almost half the profile drag. Also the maximum lift coefficient is increased significantly, its stalling characteristics however are bit more violent, but still well within safe limits. Finally, the moment coefficient is slightly less than the NACA 63_2 –415 airfoil, giving it a bit less trim drag.

More spanwise suction distributions can be found in Figures H.1 to H.6 in Appendix H. The trend with suction is a large reduction in profile drag and significant increase in maximum lift coefficient. Without suction the trend is a significant reduction in profile drag in cruising and climbing conditions and a slightly lower maximum lift coefficient than the NACA 63₂-415 airfoil.

Figure 6.6 shows the lift distribution of the new wing at cruise conditions. Table 6.2 lists the characteristics in spanwise direction in absolute numbers and Table 6.3 shows them in relative numbers. It is clear that the effects of the new airfoil and the suction are tremendous. The profile drag of the new airfoil is about 80–85% of the profile drag of the original airfoil, up to 85% span. With suction this is decreased to almost half the profile drag. Only the last 15% of the span has less improvement and the tip station has a 0.6% higher profile drag when suction is not applied.



Figure 6.5: Polars of the BW 10–144 airfoil with and without suction and the NACA 63_2 –415 airfoil without suction for $\text{Re}\sqrt{C_L} = 3.01 \cdot 10^6$ (MGC)



Figure 6.6: Lift distribution of the new wing at cruise conditions ($C_L = 0.45$)

| Station | η | $\operatorname{Re}\sqrt{C_L}$ | c_l | $c_{d \text{ NACA } 63_2-415}$ | $c_{d \text{ BW 10-144}}$ | $c_{d \text{ BW 10-144 suction}}$ |
|-----------|--------|-------------------------------|-------|--------------------------------|---------------------------|-----------------------------------|
| Root | 0.00 | $3.89\cdot 10^6$ | 0.44 | 0.004607 | 0.003690 | 0.002327 |
| Station 1 | 0.24 | $3.46\cdot 10^6$ | 0.47 | 0.004752 | 0.003844 | 0.002419 |
| Station 2 | 0.47 | $3.07\cdot 10^6$ | 0.48 | 0.004843 | 0.003989 | 0.002514 |
| Station 3 | 0.69 | $2.68\cdot 10^6$ | 0.46 | 0.004888 | 0.004055 | 0.002595 |
| Station 4 | 0.84 | $2.41\cdot 10^6$ | 0.41 | 0.004883 | 0.004073 | 0.002638 |
| Tip | 1.00 | $2.14\cdot 10^6$ | 0.15 | 0.004632 | 0.004659 | 0.003670 |

Table 6.2: The spanwise characteristics for cruise conditions $(C_L = 0.45)$

Table 6.3: The spanwise improvements for cruise conditions ($C_L = 0.45$), with the NACA 63_2 -415 airfoil as a reference

| Station | η | $c_{d \text{ NACA } 63_2-415} \ [\%]$ | $c_{d \text{ BW } 10-144} \ [\%]$ | $c_{d \text{ BW 10-144 suction}}$ [%] |
|-----------|------|---------------------------------------|-----------------------------------|---------------------------------------|
| Root | 0.00 | 100 | 80.08 | 50.51 |
| Station 1 | 0.24 | 100 | 80.89 | 50.90 |
| Station 2 | 0.47 | 100 | 82.37 | 51.91 |
| Station 3 | 0.69 | 100 | 82.95 | 53.10 |
| Station 4 | 0.84 | 100 | 83.42 | 54.03 |
| Tip | 1.00 | 100 | 100.57 | 79.23 |

Another important condition is the climb. The Eaglet has a maximum rate of climb at $C_L = 0.69$. The lift distribution of this condition is shown in Figure 6.7 and the spanwise characteristics in Table 6.4 and Table 6.5. Like at cruise condition, the drag reduction at maximum rate of climb is enormous. Without suction the new wing has about 15% less profile drag, and with suction the profile drag is halved. At higher lift coefficients the improvements mitigate, therefore a flap is installed for climbing with higher angles and for the landing, as will be explained in the next section.

6.4 Flaps Selection

When flying at high lift coefficients it is necessary to apply flaps. Flaps lower the profile drag at climb lift coefficients and they increase the maximum lift coefficient at landing conditions. The reduction in drag at climb lift coefficients is needed to get a higher angle of climb, necessarily for short field take-off and avoiding obstacles during initial climb. Also

Table 6.4: The spanwise characteristics for maximum rate of climb conditions ($C_L = 0.69$)

| Station | η | $\operatorname{Re}\sqrt{C_L}$ | c_l | $c_{d \text{ NACA } 63_2-415}$ | $c_{d \text{ BW } 10-144}$ | $c_{d \text{ BW 10-144 suction}}$ |
|-----------|--------|-------------------------------|-------|--------------------------------|----------------------------|-----------------------------------|
| Root | 0.00 | $3.89\cdot 10^6$ | 0.68 | 0.005186 | 0.004407 | 0.002448 |
| Station 1 | 0.24 | $3.46\cdot 10^6$ | 0.72 | 0.005396 | 0.004652 | 0.002567 |
| Station 2 | 0.47 | $3.07\cdot 10^6$ | 0.73 | 0.005503 | 0.004733 | 0.002660 |
| Station 3 | 0.69 | $2.68\cdot 10^6$ | 0.70 | 0.005481 | 0.004682 | 0.002741 |
| Station 4 | 0.84 | $2.41\cdot 10^6$ | 0.62 | 0.005328 | 0.004547 | 0.002759 |
| Tip | 1.00 | $2.14\cdot 10^6$ | 0.23 | 0.004689 | 0.003804 | 0.002657 |



Figure 6.7: Lift distribution of the new wing at maximum rate of climb conditions ($C_L = 0.69$)

| Station | η | $c_{d \text{ NACA } 63_2-415} \ [\%]$ | $c_{d \text{ BW } 10-144} \ [\%]$ | $c_{d \text{ BW 10-144 suction}}$ [%] |
|-----------|--------|---------------------------------------|-----------------------------------|---------------------------------------|
| Root | 0.00 | 100 | 84.98 | 47.21 |
| Station 1 | 0.24 | 100 | 86.22 | 47.57 |
| Station 2 | 0.47 | 100 | 86.01 | 48.34 |
| Station 3 | 0.69 | 100 | 85.41 | 50.00 |
| Station 4 | 0.84 | 100 | 85.35 | 51.78 |
| Tip | 1.00 | 100 | 81.13 | 56.66 |

Table 6.5: The spanwise improvements for maximum rate of climb conditions ($C_L = 0.69$), with the NACA 63_2 -415 airfoil as a reference



Figure 6.8: Various flap configurations: a) the plain flap, b) the split flap, c) the slotted flap and d) the fowler flap (Aviation Glossary [2010])

flaps are needed in an aborted landing or go-around situation to gain as much altitude as possible in a short amount of distance. In these situations, a low drag is essential and a small flap deflection is needed. In approach and landing conditions, flaps are needed to fly slowly and to lose energy before touchdown. The flaps are used to increase the maximum lift coefficient, to get a larger margin to the stall. In this situation, the flaps are deflected over a larger angle and the drag increases significantly. This increase in drag is not unwanted, during final approach the aircraft has to be decelerated as much as possible to minimize the needed landing distance.

The downside of flaps in combination with boundary layer suction is that the available chord length for suction is decreased; it is impractical to install a suction device in the flap. This is true for all types of flaps excluding the split flap. Split flaps consist of a plate which is deflected at the lower aft part of the airfoil, see Figure 6.8b. The split flap creates a lot more drag and is therefore less efficient in generating extra lift, however it leaves the upper side of the airfoil unaltered, making it ideal to extend the suction surface.

6.4.1 Split Flaps

Nowadays, split flaps are considered obsolete due to their high drag and low efficiency. However for suction, split flaps could be a viable option because no movable parts are present at the upper surface of the airfoil which disturb the flow. To investigate the

| Span: | 1.850 | [m] |
|--------------------|-------------------------------|------------|
| Inboard position: | 1.017 | [m] |
| Outboard position: | 2.867 | [m] |
| Chord: | 22.0 | [% chord] |
| Deflection: | $0\backslash 15\backslash 30$ | [° ↓] |

Table 6.6: Properties of the flaps



Figure 6.9: Correction factor for chord ratio of split flaps (Hoak [1965])

increase in maximum lift and drag coefficient, the handbook methods of Hoak [1965] and Torenbeek [1982] are used. A small flap deflection is analyzed to determine if the increase in lift is sufficient and the increase in drag is acceptable for climb conditions.

The properties of the flaps are given in Table 6.6. According to Hoak [1965], the increase in lift coefficient due to split flaps can be determined by:

$$\Delta c_l = k \left(\Delta c_l \right)_{\frac{c_f}{c} = 0.2}$$

where k is a correction factor for a flap-ratio other than $\frac{c_f}{c} = 0.2$ and $(\Delta c_l)_{\frac{c_f}{c}=0.2}$ is the increment in lift coefficient due to a split flap of flap-ratio 0.2. These two parameters can be found in Figure 6.9 and Figure 6.10 respectively. The figures are based on data sheets, corrected for aspect ratios of more than 6 and yield results within 10% of experimental data.

For a flap length of 22% chord and a deflection of 15° (climb configuration), the values of k and $(\Delta c_l)_{c_{f=0.2}}^{c_{f=0.2}}$ are 1.04 and 0.55 respectively, yielding:

$$\Delta c_l = 0.572$$



Figure 6.10: Lift coefficient increment for 20% chord split flaps (Hoak [1965])



Figure 6.11: Profile drag function $F(\delta)$ for split flaps (Torenbeek [1982])

This increase in lift coefficient is sufficient to ensure a safe margin to the stall during climb. To determine the increase in drag, Torenbeek [1982] is used. An empirical relation is given, obtained from systematic experiments (Wenzinger & Harris [1939]):

$$\Delta c_{d_p} = 0.55 \frac{c_f}{c} \left[\frac{\frac{c_f}{c}}{\left(\frac{t}{c}\right)^{\frac{3}{2}}} \right]^{\frac{2}{9}} F(\delta)$$

where $F(\delta)$ is shown in Figure 6.11. For 22% chord flaps with a 15° deflection, this will yield:

$$\Delta c_{d_n} = 0.0198$$

This increases the drag 4 to 5 times, even with boundary layer suction. This is unacceptable for take-off and climbing conditions, where all the energy is needed to gain altitude. Therefore the idea of split flaps must be disregarded. Plain flaps, which are installed on the original wing, are therefore selected, described in the next section.

6.4.2 Plain Flaps

Plain flaps are installed on the original wing of the Eaglet. They function by deflecting the aft part of the airfoil, as shown in Figure 6.8a. Because the deflection of the aft part of the airfoil does not introduce large discontinuities in the airfoil shape, the problem can be analyzed in XFOIL. Figure G.7 in Appendix G shows the potential flow pressure distribution of the BW 10–144 airfoil with 15° flap deflection.

As explained before, the laminar boundary layer is tripped to turbulent, just before the flap. This gives a 'fresh' turbulent boundary layer, which is able to cope with the adverse pressure gradient on the flap. Figure H.7 to Figure H.9 in Appendix H show the polars of three spanwise flap stations. As can be seen, the maximum lift coefficient is significantly increased and the drag is much lower compared to the calculations of the split flap. In the next section, the suction with flaps will be discussed.



(BW 10–144, $\operatorname{Re}\sqrt{C_L} = 3.07 \cdot 10^6$, $c_l = 0.93$)

6.5 Suction Distribution for Flaps

The pressure distribution with flaps, Figure G.7 in Appendix G, shows a favorable pressure gradient just in front of the flap. This favorable pressure gradient suppresses the growth of the amplification factor and the necessity for large amounts of suction, as can be seen in Figure 6.12. However, only one suction profile can be implemented on the wing and therefore the same suction is applied with flap as with the clean wing (Figure 5.5), albeit with a different intensity, as will be shown in Chapter 7. The resulting polars are also shown in Figure H.7 to Figure H.9 in Appendix H. As can be seen, the drag reduction is large, making the climb more efficient.

6.6 Suction Distribution for Ailerons

For aileron deflections, no suction distributions will be calculated. The suction system would be too complicated to account for aileron deflection. However it is investigated if the application of suction does influence the ailerons in a negative sense. Figure 6.13 and Figure 6.14 show the polars for $\text{Re} = 3 \cdot 10^6$ and Figure H.10 and Figure H.11 in Appendix H show the polars for $\text{Re} = 6 \cdot 10^6$. In those figures, the optimal suction distribution is the suction distribution shown in Figure 5.5, optimized for an aileron deflection of 15° following the criteria defined in Section 6.1. Non-optimal suction has the same shape (Figure 5.5), however its magnitude is an intermediate value between optimal and zero suction.



Figure 6.13: Polars of an aileron deflection of 15° down, without suction (red) with nonoptimal suction (blue) and optimal suction (purple).



Figure 6.14: Polars of an aileron deflection of 15° up, without suction (red) with non-optimal suction (blue) and optimal suction (purple)

As can be seen, the boundary layer suction has a positive influence on the drag and maximum lift coefficient, increasing the effectiveness of the ailerons. Also, when the suction is non-optimal, no negative situations occur.

Figure 6.14 shows an instant increase in drag at a negative aileron deflection. This is because the lower surface of the airfoil is critically designed and transition will run instantly to the nose when the laminar boundary layer cannot cope with the pressure gradient. This instant decrease in laminar flow will give an increase in drag. The drag increase is not detrimental because the relative large drag increase in absolute sense is very small on aircraft scale, moreover it will counteract the adverse yaw effect¹.

¹Adverse yaw is an aircraft yaw motion in opposite direction to the initiated roll movement, due to the increase in induced drag (due to increased angle of attack) of the up-going wing.

Chapter 7

The Final Design

7.1 The Suction Distribution

In the previous chapter the suction distribution is calculated for various spanwise stations. This is an optimal solution for these locations. Ideally, every chord would have its own suction distribution, but this is practically impossible. A choice has to be made about how many different suction stations will be needed to get a good suction result, which is technically feasible. The simplest implementation would be the same suction over the entire wing. To get laminar flow up to 75% chord, the most critical suction distribution must be choosen over the entire wing. This critical suction distribution is found at the wing tip, where the most suction is needed. This means that over the entire wing the boundary layer suction of the tip is applied.

When calculating the root station, which requires the least suction, with the suction distribution of the tip, the shape factors were checked. An example is given in Figure 7.1. As can be seen, the shape factor drops below the value of H = 2, meaning a lower shape factor than asymptotic suction. The significance of such a shape factor is yet unknown, and this option has to be disregarded. Another down-side of one suction distribution for the entire wing is that more suction is applied than necessary thus consuming more energy than needed. Also, when flaps are applied, the suction needed is much lower than without; it would be advantageous to be able to control this area separately.

Therefore the option is chosen to have two separate suction areas, one spanning the flap section and the other one spanning the aileron section. The root station is not equipped with suction for the reasons mentioned in Section 6.1. By using the same method as described above; applying critical suction over the separate suction areas, calculations are made. The shape factors stay well within the limits and with this configuration it is possible to change the amount of suction on the flap station when flaps are deflected.

Figures I.1 to I.6 in Appendix I show the suction distributions of the stations with the final suction distribution; $v_0 = -0.0519 \frac{m}{s}$ for the flap stations and $v_0 = -0.0590 \frac{m}{s}$ for the aileron stations. With 15 degrees flap deflection, the suction velocity of the flap stations becomes $v_0 = -0.0297 \frac{m}{s}$, see also Table 7.1.



Figure 7.1: A too low shape factor due to excessive suction. Displayed is the shape factor of the upper surface of the BW 10–144 airfoil at the root. Re = $3.89 \cdot 10^6$ and $c_l = 0.434$. The red lines are the boundaries; H = 2.0 is the shape factor for asymptotic suction and $\frac{x}{c} = 0.75$ the end of the suction area. The shape factor has to stay above the red line to be a physical solution for a laminar boundary layer.

 Table 7.1: Suction velocities at the various stations

| Station | $v_0 \left[\frac{\mathrm{m}}{\mathrm{s}}\right]$, Flap retracted | $v_0 \left[\frac{\mathrm{m}}{\mathrm{s}}\right]$, Flap deflected 15° |
|-----------------|---|---|
| Flap station | -0.0519 | -0.0297 |
| Aileron station | -0.0590 | -0.0590 |



Figure 7.2: The lift breakdown of the aircraft with the original wing and the new wing with suction

7.2 The Complete Aircraft

To obtain the results of the wing, XFLR has been used in cooperation with the XFOIL results. Polars of the BW 10–144 airfoil with and without suction are made and the drag coefficient is corrected according to Table 3.1 of Section 3.3. The new wing is added to the original aircraft to obtain the results of the complete aircraft.

Figure 7.2 shows the lift breakdown of the aircraft and the buildup to the new aircraft. It can be seen that the new aircraft has a slightly steeper lift curve compared to the original one. This is a bit exaggerated at higher angles of attack, because of the subtraction of the original wing, which has a lower maximum lift coefficient. In general, the lift curve does not change very much with the new wing, apart from the higher maximum lift coefficient.

Figure 7.3 shows the drag breakdown of the 'new' Eaglet. This figure is very important because it shows the improvements achieved in this thesis work. The total drag reduction of the aircraft is very small. This is because the wing profile drag is a relative small portion of the total aircraft drag. This is further clarified in Figure 7.4, the drag reduction at cruise $C_L = 0.45$ is about 3.2%.

However, when looking at the wing alone, the improvements are much more significant. Figure 7.5 shows the improvement of the isolated wing. In cruise conditions the drag reduction of the wing is about 13%, up to 20% at high flight speeds where profile drag becomes more and more significant. This demonstrates that the boundary layer suction causes a big improvement in terms of wing drag, however the large drag of the rest of the aircraft causes the total drag reduction of the aircraft to be minimal.



Figure 7.3: The drag breakdown of the aircraft with the original wing and the new wing with suction



Figure 7.4: The improvement of the aircraft with suction



Figure 7.5: The improvement of the wing with suction

Chapter 8

Conclusions and Recommendations

8.1 Conclusions

This thesis presents the design of a new wing for the Eaglet, especially designed for boundary layer suction. First was investigated how the original aircraft performs and the properties of the original wing were determined, both with a handbook method and lifting line theory by means of XFLR. A new airfoil was designed in XFOIL with boundary layer suction in mind. The new airfoil has good suction characteristics and performs very well in both suction turned-on and turned-off conditions. With the new airfoil, the new wing was formed. The twist has been removed for a less complicated implementation of the suction system. Two types of flaps were investigated for the new wing; split flaps and plain flaps. Split flaps had the advantage of a more ideal implementation of the suction system, however due to the enormous increase in drag this option had to be rejected. Plain flaps, which are installed on the original wing, were chosen and investigated in XFOIL. They have good drag characteristics during climb and are relatively simple to implement. For the wing with and without flap deflection, the suction distributions were determined for various wing stations. A choice was made to split the wing in two suction stations; one spanning the flaps and one spanning the ailerons. Also it was investigated if the suction had a negative influence on aileron deflection, which was not the case.

At this moment no satisfactory practical implementation of the suction system is available, therefore it could not be investigated how the system should be integrated in the wing. Power requirements are not calculated due to lack of detailed information about the final implementation of the system.

Finally the new wing was calculated in XFLR and these results were used to determine the performance of the total aircraft. The total aircraft showed a drag reduction of about 3.2%, which was disappointing but anticipated for due to the low aerodynamic efficiency of the aircraft, as investigated by Debrauwer [2008]. When looked at the isolated wing the results were much better, showing a drag decrease of 13% at cruise up to 20% at high flight speeds. It should be noted however, that these numbers do not include the drag equivalent to the energy costs of the suction system. After having investigated the improvement of the new wing, it is the author's opinion that the implementation of boundary layer suction in its current form is not beneficial for the Eaglet. The marginal improvements it introduces do not outweigh the highly complicated system. The Eaglet is far from an aerodynamically optimal design, and more simple improvements can be made to reduce drag. However, the Eaglet can be used as a testbed for investigating boundary layer suction. In that case, investigating boundary layer suction is the goal, not saving fuel. Some recommendations will be given in the next section.

8.2 Recommendations

After finalizing the design, the following recommendations are made:

- 1. The aerodynamic efficiency of the Eaglet should be investigated. A lot of improvements can be made by redesigning the individual parts.
- 2. When there is a practical solution for the implementation of the suction system, the real layout and off-design conditions should be investigated.
- 3. When the real layout is known, the energy consumption of the system should be calculated and processed in the final results.
- 4. To verify the results obtained by XFOIL and XFLR, the airfoil and wing should be tested in a wind tunnel.

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Appendix A

Euro-ENAER Eaglet EE–10 Data Sheet

For completeness, the Euro-ENAER EE–10 Data Sheet (Anonymous [2009]) is included in this appendix.

Design Features

- Side-by-side, two-seat monoplane;
- Composite airframe structure;
- Low wing with partial laminar flow profile;
- Fixed tricycle landing gear with nose wheel;
- Air-cooled piston engine with fixed propeller.

Power Plant

Propeller:

- 2-Blade, fixed pitch propeller with spinner;
- Material: wood;
- Manufacturer: MT-propeller;
- Type: MT-178 R160/3D.

Engine:

- Type: Lycoming O-320 D2A, rated power of engine: 160 SHP at 2700 RPM;
- Down angle: 2°;
- Right side angle: 2°.

Selected power ratings for EE–10 type:

- Max. take-off power: 160 SHP at 2700 RPM, 5 minutes;
- Max. continuous power: 145 SHP at 2450 RPM.

Dimensions External

| Total length: | 7.050 [m] | 23.13 [ft] |
|----------------------------------|-------------------|-------------------|
| Maximum height: | $2.415 [{ m m}]$ | 7.92 [ft] |
| Maximum fuselage width: | 1.220 [m] | 4.00 [ft] |
| Wing span (excluding wing tips): | 8.314 [m] | 27.28 [ft] |
| Wing span (including wing tips): | 8.700 [m] | 28.54 [ft] |
| Wheel base: | 1.472 [m] | 4.83 [ft] |
| Wheel track: | 2.800 [m] | $9.19 [{\rm ft}]$ |
| Propeller diameter: | $1.780 [{ m m}]$ | 5.84 [ft] |

Dimensions Internal

| Cabin width: | 1.120 [m] |
|--|-----------|
| Cabin height: | 1.000 [m] |
| Cabin length (bulkhead to separation plate): | 1.537 [m] |

Areas and Other Geometric Properties

For engineering purposes the datum is an artificial point in front of the aircraft. All data in this report refers to this datum in EX, EY and EZ coordinates. For example, with respect to this datum, the firewall is located at EX 1600. Published datum for operational purpose is the leading edge at the fuselage stub wing to wing joint (EX 2038, EY \pm 1020). Level attitude is defined by the orientation of the lower door frame.

| W | ing: |
|---|------|
|---|------|

| Area ¹ : | 9.85 $[m^2]$ 1 | 06.2 [ft ²] |
|---------------------------------------|-----------------|-------------------------|
| Aspect ratio ¹ : | 7.02 [-] | |
| Root $chord^1$: | 1.530 [m] | 5.02 [ft] |
| Tip chord: | 0.808 [m] | 2.65 [ft] |
| Tip $chord^1$: | 0.840 [m] | 2.76 [ft] |
| Taper ratio ¹ : | $0.549 \ [-]$ | |
| Root incidence: | 3.00 [°] | |
| Tip incidence: | 0.50 [°] | |
| MAC incidence: | 1.87 [°] | |
| Dihedral: | 5.00 [°] | |
| Sweep angle at $0.25c$: | 0.00 [°] | |
| Sweep angle at LE: | 2.38 [°] | |
| Mean geometric chord ¹ : | 1.185 [m] | 3.83 [ft] |
| Mean aerodynamic chord ¹ : | 1.218 [m] | 3.96 [ft] |
| Distance of MAC LE to datum EX^1 : | 2.073 [m] | 6.82 [ft] |
| Distance of AC to datum EX^1 : | 2.38 [m] | 7.81 [ft] |
| Distance of AC to datum EZ^1 : | 1.214 [m] | 3.98 [ft] |
| Root airfoil: | NACA 63_2-415 | |
| Tip airfoil: | NACA 63_2-415 | |

Aileron:

| Alleron: | | | | |
|-----------------------|--------|---------|------|----------|
| Area: | 0.25 | m^2] | 2.69 | $[ft^2]$ |
| Span: | 1.290 | [m] | 4.23 | [ft] |
| Mean geometric chord: | 0.194 | [m] | 0.64 | [ft] |
| Displacement: | 25.0 [| ° ↑] | | |
| | 20.0 [| °↓] | | |
| Inboard chord: | 0.213 | [m] | | |
| Outboard chord: | 0.171 | [m] | | |
| | | | | |

| F'lap: | | | |
|-----------------------|-------------------------------|---------|----------------|
| Area: | 0.49 | $[m^2]$ | $5.27 [ft^2]$ |
| Span: | 1.850 | [m] | 6.07 [ft] |
| Mean geometric chord: | 0.265 | [m] | 0.87 [ft] |
| Position: | $0\backslash 15\backslash 30$ | [° ↓] | |
| Inboard chord: | 0.298 | [m] | |
| Outboard chord: | 0.232 | [m] | |
| | | | |

¹Excluding wing tips.

| Area: | 2.13 [m ²] | $22.93 [ft^2]$ |
|---|------------------------|-----------------|
| Span: | 1.5 [m] | 4.92 [ft] |
| Aspect ratio: | 4.23 [-] | |
| Root chord: | 0.880 [m] | 2.89 [ft] |
| Tip chord: | 0.538 [m] | 1.77 [ft] |
| Taper ratio: | $0.611 \ [-]$ | |
| Root and tip incidence: | 0.00 [°] | |
| Dihedral: | 0.00 [°] | |
| Sweep angle at $0.25c$: | 5.00 [°] | |
| Mean geometric chord: | 0.710 [m] | 2.33 [ft] |
| Mean aerodynamic chord: | 0.723 [m] | 2.37 [ft] |
| Distance of AC to datum EX: | 6.240 [m] | 20.47 [ft] |
| Distance of AC to datum EY: | 0.690 [m] | 2.64 [ft] |
| Distance of AC to datum EZ: | 1.700 [m] | 5.58 [ft] |
| Horizontal distance from AC_h to AC_w : | 3.862 [m] | 12.67 [ft] |
| Root to tip airfoil: | NACA 0009 | |

Horizontal tail:

Elevator:

| Lievatori | | |
|-------------------------|------------------------------|----------------|
| Area: | $0.84 [m^2]$ | $9.04 [ft^2]$ |
| Span: | 3.000 [m] | 9.84 [ft] |
| Root chord: | 0.350 [m] | 1.15 [ft] |
| Tip chord: | 0.210 [m] | 0.69 [ft] |
| Mean geometric chord: | 0.280 [m] | 0.92 [ft] |
| Horn area: | $0.020 [m^2]$ | $0.21 [ft^2]$ |
| Displacement: | 25.0 [° \uparrow] | |
| | $18.0 [^{\circ}\downarrow]$ | |
| Position of hinge line: | 0.16 <i>c</i> | |

Vertical tail:

| Area ² : | 0.93 [m ²] | $10.04 [ft^2]$ |
|---|------------------------|-----------------|
| Span^2 : | 1.120 [m] | 3.67 [ft] |
| Aspect ratio ² : | 1.35 [-] | |
| Root $chord^2$: | 1.175 [m] | 3.85 [ft] |
| Tip $chord^2$: | 0.490 [m] | 1.61 [ft] |
| Taper ratio ² : | 0.417 [-] | |
| Sweep angle at $0.25c$: | 28.20 [°] | |
| Mean geometric $chord^2$: | 0.830 [m] | 2.72 [ft] |
| Mean aerodynamic chord ² : | 0.879 [m] | 2.89 [ft] |
| Distance of AC to datum EX^2 : | 6.591 [m] | 21.62 [ft] |
| Distance of AC to datum EZ^2 : | 2.183 [m] | 7.16 [ft] |
| Horizontal distance from AC_v to AC_w^2 : | 4.213 [m] | 13.82 [ft] |
| Root to tip airfoil: | NACA 0009 | |
| Vertical tail incidence: | 0.0 [°] | |

 $^{2}\mathrm{Does}$ not include extended area below the horizontal stabilizer.

Rudder:

| $Area^3$: | $0.415 \ [m^2]$ | $4.47 [ft^2]$ |
|-------------------------------------|--|----------------|
| Span^3 : | 1.316 [m] | 4.32 [ft] |
| Root chord ³ : | 0.439 [m] | 1.44 [ft] |
| Tip $chord^3$: | 0.192 [m] | 0.63 [ft] |
| Mean geometric chord ³ : | 0.315 [m] | 1.03 [ft] |
| Horn area: | $0.028 [m^2]$ | $0.30 [ft^2]$ |
| Displacement: | $30.0 \left[\circ \leftarrow \right]$ | |
| Fuel system: | | |

| Total volume ⁴ : | 150.0 [l] | 39.6 [gal(US)] |
|--------------------------------|-----------|-----------------|
| Unusable volume ⁴ : | 0.5 [l] | 0.13 [gal(US)] |

Selected Certification Limitations

1. Category and manoeuvres:

JAR 23, utility;

Lazy eight, chandelle, steep turn (spin to be determined).

2. Speeds please refer to:

EE-10#0200#02, Eaglet Design Speeds, ref./1/.

3. Weights and loading refer to:

EE-10#0200#03, Eaglet Weight & Balance, ref./2/.

4. Manoeuvre load factor:

4.4 positive;−2.3 negative.

5. Maximum certified altitude:

14,000 ft.

6. OAT-range:

ISA \pm 25K, resulting in: Maximum OAT at sea level, $10^{\circ}C + 25K = 40^{\circ}C$; Minimum OAT at 14,000 ft, $-13^{\circ}C - 25K = -38^{\circ}C$.

7. Certified airframe life⁵:

20,000 hours;

 $^{^{3}\}mathrm{Does}$ not include horn area.

 $^{^{4}}$ Standard tank, actual tank volume and usable fuel are subject to certification testing confirmation.

⁵The practical airframe life of the composite airframe is expected to be infinite due to the nature of the material. Nevertheless, these limits are chosen for certification substantiation/fatigue assessment.

| 30,000 | landings. |
|--------|-----------|
|--------|-----------|

8. Operational:

VFR.

Appendix B

Drawings of the Eaglet



Figure B.1: Top-view of the Eaglet (ENAER [2009])



Figure B.2: Side and front-view of the Eaglet (ENAER [2009])

Appendix C

NACA 63_2 –415 Airfoil



Figure C.1: Lift and moment coefficient of the NACA 63₂-415 airfoil, obtained by wind tunnel testing (Abbott & Von Doenhoff [1959])



Figure C.2: Drag and moment coefficient of the NACA 632-415 airfoil, obtained by wind tunnel testing (Abbott & Von Doenhoff [1959])

Appendix D

Comparison Calibrated XFOIL with Abbott & Von Doenhoff



Figure D.1: Calibrated XFOIL and Abbott & Von Doenhoff for $\mathrm{Re}=3\cdot10^6$



Figure D.2: Calibrated XFOIL and Abbott & Von Doenhoff for $\mathrm{Re} = 6\cdot 10^6$



Figure D.3: Calibrated XFOIL and Abbott & Von Doenhoff for $\mathrm{Re}=9\cdot10^6$



Figure D.4: Calibrated XFOIL and Abbott & Von Doenhoff for ${\rm Re}=3\cdot 10^6$



Figure D.5: Calibrated XFOIL and Abbott & Von Doenhoff for $\mathrm{Re}=6\cdot 10^6$



Figure D.6: Calibrated XFOIL and Abbott & Von Doenhoff for $\mathrm{Re}=9\cdot 10^6$

Appendix E

Lift Distributions for the Original Wing







Figure E.2: Lift distribution for $C_L = 0.4$







Figure E.4: Lift distribution for $C_L = 0.8$



Figure E.5: Lift distribution for $C_L = 1.0$



Figure E.6: Lift distribution for $C_L = 1.2$

Appendix F

Results of XFLR







Figure F.2: The induced drag coefficient of the original wing



Figure F.3: The profile drag coefficient of the original wing



Figure F.4: The total drag coefficient of the original wing



Figure F.5: The lift coefficient of the new wing (without suction)



Figure F.6: The induced drag coefficient of the new wing (without suction)



Figure F.7: The profile drag coefficient of the new wing (without suction)



Figure F.8: The total drag coefficient of the new wing (without suction)

Appendix G

Wortmann FX S 03–182, FX 38–153 and the BW 10–144 Airfoil







Figure G.2: Potential flow pressure distribution of the FX S 03-182 airfoil







Figure G.4: Potential flow pressure distribution of the FX 38-153 airfoil







Figure G.6: Potential flow pressure distribution of the BW 10-144 airfoil



Figure G.7: Potential flow pressure distribution of the BW 10–144 airfoil with 15° flap deflection

Appendix H

Suction Distribution at Various Spanwise Stations







Figure H.2: Polars of Station 1


Figure H.3: Polars of Station 2



Figure H.4: Polars of Station 3







Figure H.6: Polars of the tip



Figure H.7: Polars of Station 1, with 15° flap deflection



Figure H.8: Polars of Station 2, with 15° flap deflection



Figure H.9: Polars of Station 3, with 15° flap deflection



Figure H.10: Polars of an aileron deflection of 15° down, without suction (red) with nonoptimal suction (blue) and optimal suction (purple)



Figure H.11: Polars of an aileron deflection of 15° up, without suction (red) with nonoptimal suction (blue) and optimal suction (purple)

Appendix I

Suction Distributions of the Final Design



Figure 1.1: Polars of Station 1, with non optimal suction



Figure 1.2: Polars of Station 2, with non optimal suction



Figure 1.3: Polars of Station 3, with optimal suction



Figure I.4: Polars of Station 3, with non optimal suction



Figure 1.5: Polars of Station 4, with non optimal suction



Figure 1.6: Polars of the tip, with optimal suction