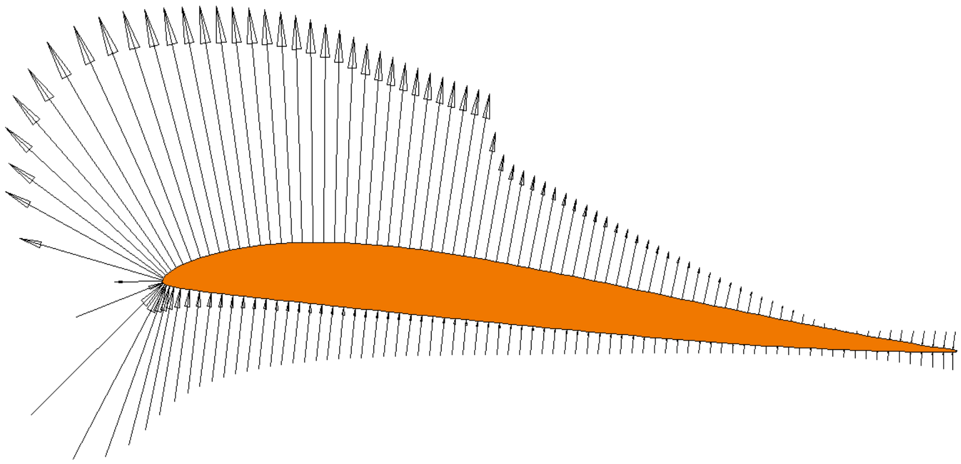
School of Innovation, Design and Engineering

BACHELOR THESIS IN
AERONAUTICAL ENGINEERING
15 CREDITS, BASIC LEVEL 300

Project Solaris – Analysis of airfoil for solar powered flying wing UAV



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Report code: MDH.IDT.FLYG.21.10.2011.GN300.15HP.Ae

Abstract

This study is part of the second phase of the Solaris project, where the aim is to develop a solar powered Unmanned Aerial Vehicle (UAV). The second phase involves the design and optimization of the aircraft. One of the important focuses in this phase is the determination of the airfoils shape. This report sole objective is to determine which airfoil that is best suited for the aircraft, as well as presenting the airfoils characteristic properties, in comparison to other similarly airfoils.
This analysis has been carried out using XFOIL, an airfoil analysis tool developed by the MIT professor Mark Drela.

What has been done in this report:

- Comparison between a number of potential airfoils.
- Determination of the winner airfoil.
- Comparison between the winner airfoil and a conventional (non-reflexed trailing edge) airfoil.
- Calculation of the hinge moments on the winner airfoil for different flap settings.

Winner airfoil:

The Phoenix (Phönix) turned out to be the best airfoil in the comparison, closely followed by the S5020 and the S5010. Phoenix had the highest value of the parameter sought to optimize, which is endurance ($C_L^{3/2} / C_D$).

Phoenix maximum endurance ($C_L^{3/2} / C_D$) for five different Reynold numbers:

Re	Endurance
400.000	85,64
300.000	78,26
200.000	67,74
100.000	49,47
50.000	25,52

Phoenix geometry:

- Maximum thickness (in percentage of chord): 8,194%
- Maximum camber (in percentage of chord): 2,774%

Date: 21 OCT, 2011
Carried out at: (Mälardalen University Sweden)
Advisor at MDH: Gustaf Enebog
Examiner: Gustaf Enebog

Sammanfattning

Den här rapporten är en del i den andra fasen av Solaris-projektet, där målet är att utveckla en solcellsdriven obemannad flygande farkost (Unmanned Aerial Vehicle, UAV). Den andra fasen omfattar konstruktionen och optimeringen av farkosten (flygplanet). En av de viktigaste byggstenarna i den här fasen är fastställandet av vingprofilens form och design, det vill säga valet av vingprofil. Målet med den här rapporten är att helt och hållet bestämma vilken vingprofil som ska användas på flygplanet, samt en presentation av dess karakteristiska egenskaper. Detta har gjorts i jämförelse med andra liknande vingprofiler. Verktyget som använts för denna jämförelse heter XFOIL. Programmet är utvecklat av professor Mark Drela på MIT i USA, och är gjort för analysering av vingprofiler.

Det här har gjorts i rapporten:

- Jämförelse mellan en mängd olika potentiella vingprofiler.
- En vinnarprofil har valts ut.
- Jämförelse mellan vinnarprofilen och en vanlig (non-reflexed trailing edge) vingprofil.
- Beräkning av de moment som krävs för att styra klaffar och roder.

Vinnarprofilen:

I jämförelsen visade det sig att Phoenix var den bästa vingprofilen, tätt följt av S5020 och S5010. Phoenix hade det högsta och bäst värdet i den eftersökta parametern ”endurance” ($C_L^{3/2}/ C_D$).

Phoenix maximum endurance ($C_L^{3/2}/ C_D$) för fem olika Reynoldstal:

Re	Endurance
400.000	85,64
300.000	78,26
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100.000	49,47
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Phoenix geometri:

- Maximal tjocklek (i procent av kordan): 8,194%
- Maximal välvning (i procent av kordan): 2,774%

Datum: 21 OKT, 2011
Utfört vid: (Mälardalens Högskola, Sverige)
Handledare på MDH: Gustaf Enebog
Examinator: Gustaf Enebog

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Tables and figures

1 Nomenclature

Symbol	Explanation
α	Angle of attack
C_D	Drag coefficient
C_{hm}	Hinge moment coefficient
C_L	Lift coefficient
D	Drag force
L	Lift force
μ	Absolute dynamic viscosity coefficient
p	Ambient air pressure
q	Dynamic pressure
Re	Reynolds number
ρ	Ambient air density
S	Wing area
V	Freestream velocity

2 Equations and formulas

Formula	Explanation	Number
$D\ V$	This is proportional to the power required to obtain level flight for a propeller driven vehicle.	(1)
$D = C_D\ \rho\ V^2\ S$	Drag force.	(2)
$L = C_L\ \rho\ V^2\ S$	Lift force.	(3)
C_L/C_D	Glide ratio, equal to formula (5).	(4)
L/D	Glide ratio, equal to formula (4).	(5)
$(C_L^{3/2} / C_D)_{max}$	State where maximum endurance is found, equal to formula (7).	(6)
$(L^{3/2}/D)_{max}$	State where maximum endurance is found, equal to formula (6).	(7)
$Re = \rho\ V\ c\ /\ \mu$	Reynolds number.	(8)

3 Background

Solaris is a student project led by Gustaf Enebog at Mälardalens University, Västerås, Sweden. The aim of the project is to design a solar powered flying wing unmanned aerial vehicle (UAV). The project takes all phases of aircraft development into reality.
The phases are divided into different stages. It's in these stages that the many participating students all have different tasks and goals to achieve before the project can finally be completed.

Together the students will form an aircraft that will fly completely with the energy from the radiating sun. The photovoltaics placed on top of Solaris will inhale the energy from the sun and convert it into electricity that can supply the electric motors of the aircraft, and give it the power needed to soar up into the skies. Solaris is divided into four big phases.

The first phase is the stage where the basic designing of the aircraft is done, as well as studies to see if the project can actually be realized. The next phase is the stage where the design is optimized and eventually finalized, which is the point where this bachelors thesis come in. This is done with precise calculations that finally will end up with the aircrafts exact shape, design and performance data.

Determination of the airfoils shape might seem to be a very small detail in the phase, however it is one of the very key factors to get right in order for the energy balance to be positive. When the determination of the airfoil is done, the aircraft will take its shape in SolidWorks CAD.

The third phase is the actual construction of the first prototype Solaris. The fourth and last phase is the stage where Solaris is further tested and optimized, especially in terms of navigation and control aspects. This phase also allows modifications, testing of different kind of payloads such as cameras and sensors. Making changes to expand the areas application of the aircraft can be done in this phase, as well as development of the interaction between the aircraft and the equipment it carries.

4 Intention

The objective of this work is to find a suitable airfoil for Solaris and determinate its exact shape (including potential modifications in thickness, camber etc.), coordinates and performance data. To complete this task it is needed to learn more about airfoils in general and airfoils for flying wings in particular. To begin with it is important to learn how different airfoils behave differently at low Reynold numbers.

Because of Solaris being a tailless flying wing, an airfoil with a reflexed trailing edge will have to be used. The already existing field of reflexed airfoils is relatively small, and needs to be examined thoroughly to find the most suitable one for this project. Solaris is an unsweept flying wing with a constant chord over its whole span. This is called a plank flying wing.

Since the aim is to fly Solaris at minimum cost of energy, the evaluation process will be done in a way where we can find the one with the best endurance characteristics instead of the more common objective of maximizing range. Before beginning the evaluation process a pool of reflexed airfoils will have to be gathered. To evaluate which airfoil actually is the best one for this project, the interactive program XFOIL will be used.

XFOIL is a tool for “designing and analysing subsonic isolated airfoils”. With XFOIL the user is able to do good, easy and reliable two dimensional analysis of the flow around an airfoil. The program fits the Solaris project very well and makes the method used in the evaluation process fairly easy compared to other programs. Aero-data shall be presented in the report by the results of XFOIL. Calculation and design of the hinge moments for the control surfaces of the aircraft shall also be included and presented in this report.

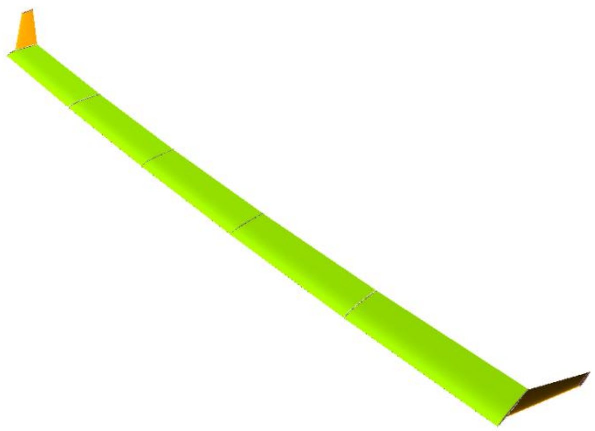

Figure 1. Example image of current but tentative design (five wing panel version). Motors on leading edge is not modeled. (Generated by Gustaf Enebog in SUMO).

5 Method

5.1 Thoughts

The main goal and aim is to fly Solaris by using the lowest possible power required from the propeller driven electric motors. This is of course at the expense of other factors, such as speed and range.

Since the wing has no aerodynamic or geometric twist and it also has a relatively high aspect ratio (induced drag is a very low portion of total drag), the results from a two dimensional simulation in our case will in theory end up with a more similar results to a three dimensional simulation typically would result in. Running a two dimensional simulation is therefore an obvious choice for us, since it considerably simplifies the analysis.

Solaris is a flying wing with no sweep. Swept wings can obtain longitudinal stability thanks to its wingtips being behind its nose thus creating a lever arm in which a canceling moment can be created, along the planes length. On an aircraft with swept wings, when the angle of attack increases, the vehicle's tip-moment will also increase thanks to its shape, and try to force the aircraft back to its horizontal level of flight. In the same manner when the angle of attack decreases and goes negative, a moment in the opposite direction will be induced trying to force the vehicle back to its previous state. When the aircraft itself continuously strives to achieve this state, it is called that the aircraft is longitudinal stable. When lift is created on any conventional cambered wing, a positive moment will always be created, and try to tip the leading edge downwards (into a dive). This is what the conventional stabilizer prevents from happening, when it creates a negative moment (in the other direction) and keeps the whole aircraft in the right position.

A flying wing with no sweep is usually called a "plank" flying wing or simply an upswept flying wing. The longitudinal stability and moment around the center of gravity is completely determined by the shape of the airfoil itself.
This is achieved by giving the airfoil a very special shape. The trailing edge is given a reflexed trailing edge. By curving the trailing edge upward, the negative moment on the airfoil can be reduced, and neutralized. It is even possible to give the airfoil a total moment that is slightly positive.

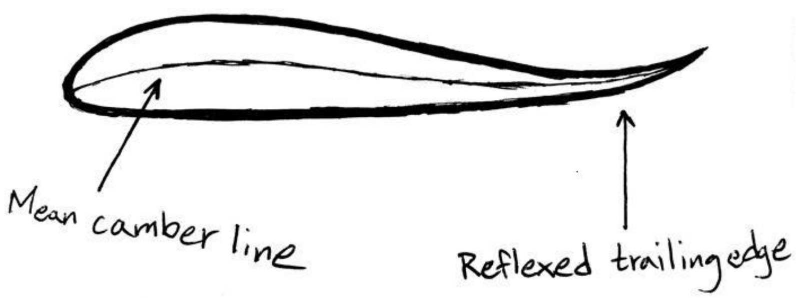

Figure 2. The image illustrates the typical characteristic shape of an airfoil with a reflexed trailing edge.

Obviously, these changes come with disadvantages. It reduced the lift coefficient and the maximum lift coefficient. These can be compensated for by increasing the camber, but this has negative effects on the moment coefficient, which required the reflexed trailing edge in the first place! We want the moment coefficient to be close to zero, or slightly above zero.

By moving the maximum camber towards the leading edge of the airfoil, we can even out some of the lost positive moment. We end up with an airfoil with an S-shape (positive camber in forward portion and negative camber in aft portion of airfoil), looking like anything but a conventional airfoil. This new airfoil is because of its shape, very sensitive to laminar separation close to the leading edge, especially at low Reynolds numbers. Solaris is being designed for low Reynolds numbers and this is where the real challenge enters in. The goal is to find the perfect airfoil for our goal. Balancing these factors back and forth is the only way to go.

Choosing the best airfoil for an airplane can be done in many different ways and with many different tools.
As already explained in the beginning of the method-chapter, a two dimensional simulation in our case is a very good approximation of reality. Hence, a two dimensional simulation is the obvious choice of method. The perfect program for this kind of task is XFOIL.
XFOIL is a program developed by Mark Drela from the MIT in the 1980's. It's written in the programming language FORTRAN and its interface consists of menu driven routines led by the user's commands. With XFOIL the user has the ability to test different airfoils at different angles of attack, Reynold numbers and Mach numbers. XFOIL gives information about lift, drag and moment coefficients, as well as pressure and velocity distributions, with many other things.

5.2 Theory - Finding the best endurance

For a propeller driven vehicle, the powered required is proportional to:

$$\mathbf{D \bullet V} \qquad (1)$$

(where D stands for drag and V stands for velocity)

For straight and level flight where $\mathbf{C_L}$ is constant, $\mathbf{C_D}$ is inversely proportional to $\mathbf{C_L/C_D}$. It is very clear that the smaller $\mathbf{C_D}$ is, the greater $\mathbf{C_L/C_D}$.
From the following common formula we can clearly see that **V** varies in square with the lift coefficient (assuming that the other parameters are constant).

$$\mathbf{L = C_L \, \rho \, V^2 \, S} \qquad (2)$$

(where L is the Lift, C_L is the lift coefficient, ρ is the air density, and S is the wing area.)

Therefore **V** is inversely proportional to $\sqrt{\mathbf{C_L}}$. What do we then get out of this? Since $\mathbf{C_D}$ is inversely proportional to $\mathbf{C_L/C_D}$ and **V** is inversely proportional to $\sqrt{\mathbf{C_L}}$, the powered required (**D • V**) is inversely proportional to the product of $\mathbf{C_L/C_D}$ and $\sqrt{\mathbf{C_L}}$ which is the same as $\mathbf{C_L^{3/2}/C_D}$ (4).
(Source: *Mechanics of flight, 11th edition, page 173.*)

The conclusion is: The greater value on $\mathbf{C_L^{3/2}/C_D}$, the smaller value on **D • V**. That means, the greater value we can find on $\mathbf{C_L^{3/2}/C_D}$, the less power is required to achieve constant level of flight.

The highest glide ratio (lift to drag ratio) that yields the best range is easily found graphically in a $\mathbf{C_L}$ **vs** $\mathbf{C_D}$-graph. It is given by drawing a linear line (thin straight line in figure X) from origo, just touching the C_L vs. C_D-curves top part, creating a tangent to the curve. The point where the two lines meet represents the values on C_L and C_D where maximum glide ratio is achieved (the red square in figure 3).

However, the best endurance is as already explained, given by the maximum $\mathbf{C_L^{3/2}/C_D}$ (equal to $\mathbf{L^{3/2}/D}$). This is found further up on the drag polar-curve (the green circle in figure X) Maybe you want to comment on this point in the graph like if there is a graphical way of finding it (I cant remember that there is. It means that the best endurance is found on a higher C_L and C_D than the values for best glide ratio. The ideal shape of the C_L vs. C_D-curve would be staying as close to the thin linear line (in figure X) as possible, without crossing it. Crossing this line would create a new $\mathbf{(L/D)_{max}}$ which is not what we want since maximizing this parameter as well most certainly negatively affect the parameter that we DO want to maximize namely $\mathbf{C_L^{3/2}/C_D}$. Staying just beneath this line would give us the highest value on $\mathbf{(L^{3/2}/D)_{max}}$. The shape in the upper left region of the drag polar-curve is therefore of highest importance when comparing airfoils with the purpose to find the best endurance characteristics. What we want is a steep drag polar-curve, high up to the left, close to the thin linear line.

The ideal drag-curve

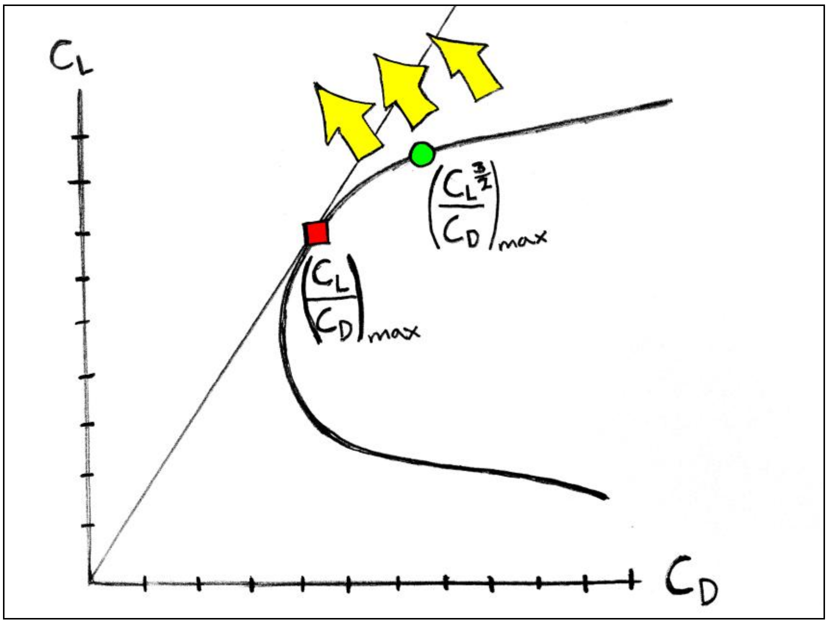

Figure 3. Illustration of how the ideal drag-curve should look like, when the aim is to maximize endurance.

5.3 Limitations

XFOIL is limited to two dimensional simulation and analysis. No experimental validations are done in this study. Hence, the separation points are all fairly precise estimations, as well as the momentum (which is dependent of the separation point is). Since Solaris is an unsweept flying wing with a constant chord and a very high aspect ratio, the calculated data is very close to reality. The induced drag is a quite small portion of the total drag, and can be calculated with a relatively high precision.

5.4 Design approach, Step by step.

The image below shows the steps in the working process. We will discuss and explain the steps below.

Step 1

Gather a quantity of potential reflexed airfoils. Some not optimized for plank flying wings (some are designed for swept wings). But these airfoils might still bring valuable properties. It is therefore a good idea to bring these for further testing in learning purposes.

Step 2

XFOIL analysis for all airfoils. They are all tested within a span on values on Reynold numbers. This is a rough and quick way to get a reasonable picture of the different designs. This process helps us to weed out and eliminate airfoils that are clearly not suitable for our goal.

Step 3

The now smaller quantities of potential airfoils are tested to evaluate which is the best one.

Step 4

Hinge moments are being calculated on the chosen airfoil. This is done with two different sizes on the flaps, with different amounts of deflection.

Step 5

Comparison of the winner airfoil and an ordinary high lift airfoil (non-reflexed). The comparison airfoil is designed for a conventional aircraft with tail (due to its positive moment). The comparison shows how much is lost in aerodynamic strength for choosing a reflexed airfoil. It brings the question; Is it worth using a tailless aircraft (flying wing) for this purpose?

Figure 4. The general steps in the working process.

• **Step 1**

Because of reflexed airfoils being relatively rare it is hard to find a greater quantity of these. The most well-known and widely used airfoils have been built and designed by model airplane enthusiasts. Reflexed airfoils have rarely been used on larger airplanes. One example is the modified Liebeck airfoil called LA2573A that was used on the NASA Pathfinder project. In an early search for potential candidate reflexed airfoils this airfoil was tested as well, but it was quickly shown that the airfoil was way too thick for our purpose and low Reynolds numbers. This was the case due to laminar separation and disturbance in the boundary layer. The phenomenon is more common on reflexed airfoils because of the simple reason that they are more cambered towards the leading edge than other ordinary airfoils. Hence, the risk of bubbles is a lot larger on the top surface on reflexed airfoils then on conventional alike. This is because they have a harder time to redirect the air at this sensitive point, especially at low Reynolds numbers.

Some of the airfoils in the pool that was gathered might not be any obvious candidates for becoming a winner airfoil, but still contribute with something during the evaluation process. Some in the way of more understanding and knowledge of what is good and what is not so good, when it comes to designing airfoils for this purpose. It also gives information of what characteristics to look for when doing the XFOIL analysis, as well as giving learning experience in the program.

• Step 2

All airfoils are tested within a span of angel of attack, at different Reynold numbers . The main goal in this step is to find the airfoils that stand out when it comes to endurance. The testing is being done by making analysis for each and every airfoil at five different Reynold numbers (the second lowest one can be considered as the critical Reynold number, and the lowest one can be seen as a way to give more understanding to the mechanics and behaviour of the Reynold number itself).The tests are made at an interval of angle of attack. The analysis in XFOIL output data for lift and drag coefficients and is being saved into a text file (in columns and rows). The airfoils with the best characteristics for maximizing $(L^{3/2}/D)_{max}$ can be easily found by this method. Output data from the text file is being put into a spreadsheet in Excel, and simply calculated by the formula. The airfoils with the best characteristics (highest values on $(L^{3/2}/D)_{max}$) have that good and steep curve that was explained in the introduction, which is what we are looking for. At this point a lot of airfoils will be eliminated. A smaller quantity of airfoils, those that really stand out when it comes to endurance, will be allowed to continue to step 3, the next qualification round.

• Step 3

Comparison of the six best airfoils is done in this step. Comparing the values on a wider range of angle of attack will give the best airfoil for Solaris. Some airfoils with close to the same maximum endurance can still have a large variation on this on this parameter at ~~a~~ different angles of attack. An airfoil with a higher endurance at a larger width on angle of attack is of course the favoured airfoil. The conclusion is; if two given airfoils have the same peak endurance, the one with the greater width (the flatter optima) on endurance is the better one.

• Step 4

It is important to know the hinge moments of the flaps when designing an aircraft. It gives the information needed to calculate the force that is required (when the flap has been sized) to deflect the control surface on the wing. This is good to know when sizing the actuators (servos). For this you need to know strength required from the actuators (you don't want to have over dimensioned actuators in an aircraft where the weight is such a critical aspect at the same time as an actuator failure always is critical on an aircraft.

Hinge moments are calculated in XFOIL after the flaps have been created inside the program. The flaps are made inside the geometric design menu. All versions with all the different angles of deflection will be saved as an entire airfoil, just like any other airfoil in XFOIL. First when that is done, every airfoil (they all have different angle of attacks and flap size) will be tested in the operate menu. They are all tested under the condition that requires the highest force to deflect the flaps. These conditions are at the highest Reynold number which is 400.000, and at the highest guesstimated flying speed which is 15m/s.

Flaps will be made in two different sizes. One will be sized to 10% of the cord length and one will be sized to 20% of the cord length (these are relatively small but are believed to maybe be sufficiently). The axis of the flap will be attached on the top surface of the wing. Both designs will be studied in XFOIL to find the hinge moments. This includes both zero flap deflection, as well as the hinge moments for maximum flap deflection in the extreme positions of the control surface. The maximum deflection positions are set to 50 degrees downwards and 20 degrees upwards. It's in the extreme positions the powered required for deflecting the control surface is at its maximum.

• Step 5

Comparison with an ordinary airfoil is very interesting in the sense of how much is lost in aerodynamic strength for choosing a reflexed airfoil. The basic design of Solaris demands an airfoil, where the moment around the centre of gravity is close to zero. Any conventional aircraft that is build with a tail and a stabilizer uses an airfoil with a positive moment around the centre of gravity. Hence this is why a conventional aircraft have a stabilizer that brings the same magnitude of cancelling moment. Comparison between the Solaris winner airfoil (reflexed) and an ordinary high lift airfoil shall be done to see how much is lost in terms of aerodynamic strength from different aspects. The parameters that will be compared is the glide ratio, the maximum lift coefficient, and of course maximum endurance.

5.5 Learning XFOIL

Learning XFOIL is a big part of the working process in this project. Knowing a lot of aerodynamic theory before using the program is a requirement but not enough. XFOIL is good for its simplicity and short commands, but the program takes a lot of time to learn for inexperienced users. The program is definitely not self-learning, and requires a lot of perquisites in terms of aerodynamics. Great help can be found on internet discussion forums, and/or by asking people with experience in the program. The user-manual is good and quite extensive, but it is far from enough to suffice all the questions and problems that occur when working in this simple but still complex program.

5.6 Using XFOIL

5.6.1 Basics

There are different ways to work with airfoils in XFOIL. When creating completely new airfoils in XFOIL, the method and menus are called: *Full-inverse surface speed design routine* (**QDES**) and *Mixed-inverse surface speed design routine* (**MDES**). When using this method, the user inputs an entire specification of the speed distribution (**Qspec**) over a whole airfoil inside (**QDES**). This in turn is worked with, to eventually generate a completely new geometry for an airfoil. The **MDES** is needed when the speed distribution (**Qspec**) needs certain modifications. If modified inside the **MDES**, the new speed distribution will again be the input data used in the **QDES**. This design method and menus of XFOIL are not used in this work. The method of choice is to compare already existing airfoils, and evaluate which one of them is the best suited one for this project. This is done by nalysed them inside the *analysis routine* menu (**OPER**), and modifying them inside the *geometric design routine* menu (**GDES**).

Before working in XFOIL, all airfoil coordinates for the possible airfoils are downloaded and saved into .dat files.

The first command that needs to be used if you want to start working with an airfoil is the load command. For example:

> LOAD c:/airfoils/common/naca4412.dat

This will make the chosen airfoil our buffer airfoil, and ready for the next step of the evaluation inside XFOIL, the smoothing of the panelling nodes.

XFOIL data flow

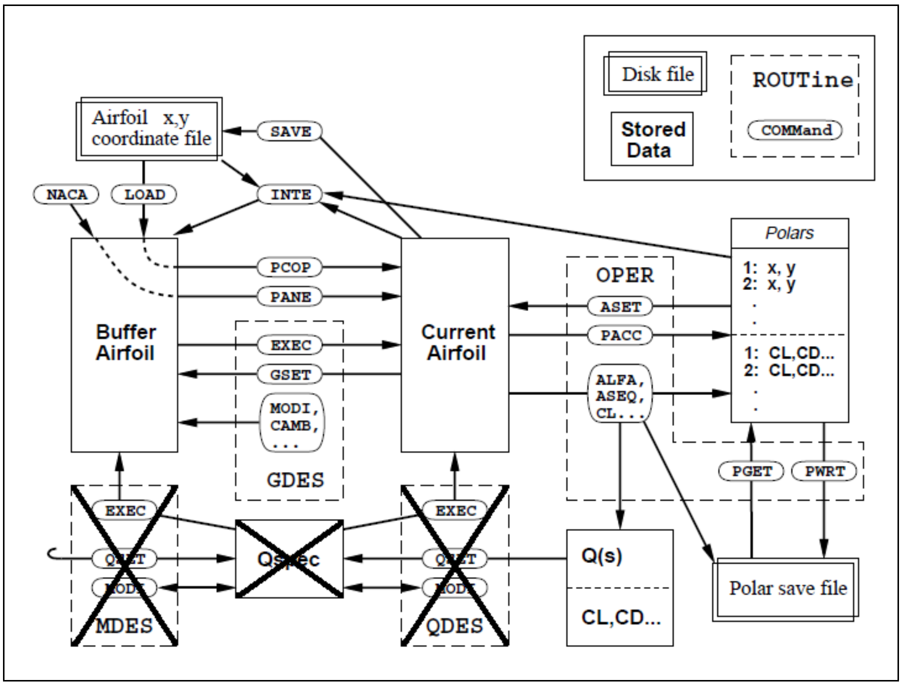

Figure 5. Illustration of the data flow in XFOIL. The menus marked with a cross are not used in the Solaris design method. The illustration was originally taken from "*XFOIL 6.9 User guide, Mark Drela, MIT Aero and Astro. Harold Youngren, Aerocraft, Inc. 11 Jan 2001 (page 7)"*.

5.6.2 Smoothing

The standard procedure that is being done before working with an airfoil is to increase the number of panelling nodes. This is done to smoothen out the line that forms the airfoil surface and edges. When loading coordinates from an airfoil into XFOIL the program draws a line between all the nodes, and together all of these create the shape of the entire airfoil. The difference in angles between two of these lines (that are connected together with a node), can, if the coordinates are too few, create a very sharp edge, with the node being inside the corner (specially close the leading and trailing edge where these angles and changing of the curve is big). These edges should not exceed an angle of about 10 degrees. Too sharp edges give problems for XFOIL to calculate the flow which in turn can result in a failure in convergence of the data. If the flow fails to converge at any point, no more flow can be calculated and analysed behind that point in direction of the streamline.
By adding nodes between all the existing nodes and by smoothing out the lines with the **PANEL** command, the risk of XFOIL failing to converge is heavily decreased.

The schematic pictures below illustrate how adding nodes can reduce the sharpness of the edges that are created when connecting one node to another. Consider **Θ** being an angle between two lines that are connected with a node. The difference in degrees between the lines should never exceed, as already said, an angle of about 10 degrees.
This means that **Θ** (the outside and bigger angle created around a node) should be no greater than 180 degrees plus the 10 degree margin that XFOIL demands. Hence, **Θ** should not exceed an angle of about 190 degrees.

Now consider $\boldsymbol{\Theta_2}$ being within the span of values where XFOIL can successfully converge the data.
In ***Figure 4*** the angle called $\boldsymbol{\Theta_1}$ is bigger than $\boldsymbol{\Theta_2}$ and anything of what XFOIL can handle. ***Figure 5*** shows how adding a node lowers the value on $\boldsymbol{\Theta}_1$. This makes the line that forms the shape of the airfoil a lot smoother.

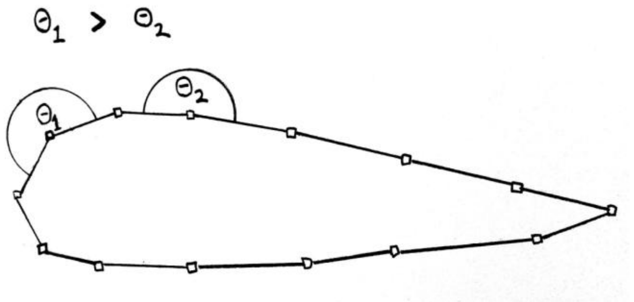

Figure 6. Illustration of how lack of nodes can create too big angles between the connecting lines.

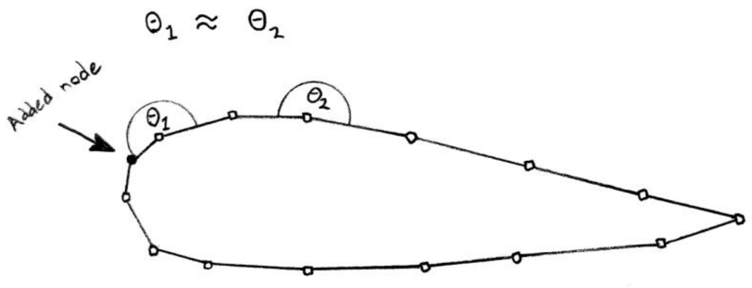

Figure 7. Illustration of how adding a node decreases the angle between the connecting lines.

The command for adding nodes is simply found and done in the panelling parameters menu. For example, if the desire is to increase the panelling nodes from the standard value of 160 to a new higher value of 200 nodes, the following series of commands should be typed in XFOIL:

> **PPAR**
>
> **n “200”**
>
> **[ENTER]** (to get to the main menu)
>
> **PANEL**

5.6.3 Modifying the airfoil

Often XFOIL have problems generating data because of an uneven distribution of the centre of pressure. Making modifications on the panel can be useful at these times. It is done inside the geometric design menu:

> **GDES**

After entering this menu, it is possible to zoom in and out at any chosen spot of the airfoil.

> **ZOOM** (by marking the wanted area to enlarge in the graphical interface window)
>
> **UNZOOM**
>
> **MOVEP** (allows the use to move points in the graphical interface window)
>
> **ADDP** (adds a point, this is also done in the graphical interface window)
>
> **DELP** (deletes a point, this is also done in the graphical interface window)

Modifications of the contour can also be done with the cursor by plotting the shape of the desired change. It’s done with the modification command:

> **MODI**

Once the command is typed, modifications can be done inside the graphical interface window. The following commands can be used while in this state:

> **A** (*ABORT* – aborts the modification process of the contour)
>
> **E** (*ERASE* – erases the most recent modifications in order)
>
> **D** (*DONE* – completes the modification)

There are a few helpful commands, that simplifies the task:

> **I** (*IN* – zooms in at any chosen spot inside the graphical interface window)
>
> **O** (*OUT* – zooms out at any chosen spot inside the graphical interface window)
>
> **P** (*PAN* – pan with the cursor)

5.6.4 Creating flaps

Generating flaps in XFOIL is easy, but making calculations needs some working with the placement and spacing between some of the nodes that forms the panel of the airfoil. Setting a new flap (deflecting the airfoil) is done under the geometric design menu. For example, setting flaps at 80% of chord length (counting from the leading edge) and placing the axis for the flap on the top side surface of the airfoil (which in our case on the phoenix is at 1.76% of the chord length in the Y-direction (counting from the chord line and upwards), and with a deflection of 50 degrees downwards, we input:

> **GDES**

> **FLAP "0.8" "0.017" "50"**

Once the flap is created inside the geometric design menu, the flapped airfoil is saved, by typing:

> **SAVE "filename"**

For XFOIL the flapped airfoil is just like any other airfoil, but with a shape that looks like an airfoil with a flap. That means that we now have created a solely new airfoil. Every angle and size of flap needs to be saved like any other airfoil. Every version represents our ~~winner~~ airfoil with a flap deflected at a certain angle.
The flapped airfoil is still a buffer airfoil. To make it the current airfoil that we want to analyse inside the analysis routine menu (**OPER**), it needs to be executed with the following command:

> **EXEC**

5.6.5 Generating hinge moments

Typing enter one time will get us out of the geometric design menu, and into the main menu of XFOIL. Once there, the panelling procedure can be good to do once again, since the airfoil has a completely new shape and all the nodes have changed positions. When the airfoil looks okay, the operate command shall be typed to enter the menu where we run the analysis:

> **OPER**

Once inside the operate menu, we specify the conditions, such as viscosity (Reynolds number), freesteam speed (Mach), and angle of attack. When that is done we need to specify where we want to calculate the hinge moment on the airfoil. To let XFOIL know this is a new flap, we type the command:

> **FNEW**

Nothing will happen until we actually tell XFOIL that we want to calculate the hinge moment. When we give that command, XFOIL will request a specification of the location of the hinge. The command for calculating the hinge moment is:

> **FMOM**

This will make XFOIL request both the X-coordinate and the Y-coordinate for the location of the hinge. Since the reflexed airfoils are relatively flat close to the trailing edge, we can take the coordinates found inside current airfoils' .dat-file, to specify the location. XFOIL will request the coordinates:

> **Enter flap hinge X location r> "number"**
> **Enter flap hinge Y location r> "number"**

After typing enter; the hinge moment coefficient is displayed. An often occurring problem can be XFOIL failing to converge the data, due to the complex shape of a reflexed airfoil, which is also flapped with an extreme deflection angle. At these times it can be useful to go back to the **PANEL** and **PPAR** menu, to check the nodes and angles once again, so that no angles and corners are too sharp. Taking small steps from low angles of attack up to the desired angle is something needed for most flapped airfoil. If instantly specifying a large angle of attack, XFOIL will most likely fail. Every step when increasing the angle of attack, initializes a new boundary layer and makes the next calculation at an even greater angle of attack more likely to be successful. For every calculation, the program uses a "Newton solution method" to calculate the data for that specific case. The newton solution for a new alpha will base its calculations on the last successful solution. The method is more likely to be successful if the difference between the previous and the new solution is as small as possible.

5.6.6 Generating polar plots

When an airfoil is going to be analysed in XFOIL, it first needs to be ready in terms of placement of the nodes that create its shape. When the panel has been smoothed out and the airfoil looks good, without any edges that are too sharp, it can be analysed and tested. The testing is done inside the analysis routine menu, which opens with the **OPER** command. Before doing any analysis in XFOIL we need to specify the conditions to XFOIL. What first needs to be done is to tell XFOIL that we want viscous flow, which is set by typing:

> **VISC**

The second thing that needs to be specified, is the Reynold number. It's done by typing the command:

> **RE "number"**

Another thing that needs to be specified, is of course the freestream speed, which is done in terms of the non-dimensional Mach number. The speed is connected to the Reynold number, because the Reynold number varies with the freestream speed. Hence, each of the guesstimated speeds used in the analysis, are connected to the calculated Reynold number. We tell XFOIL what speed we want for that specific Reynold number by typing:

> **MACH "number"**

Once that is done, we need to specify the critical amplification exponent for the turbulence of the ambient air (n_{crit}). The value on this parameter is the logarithmic exponent of the factor that triggers disturbance and turbulence. Different values on this number can represent different environments at which the airfoil will operate in. A value of 9 will be used, which can be compared to the freestream air quality and turbulence of an average wind tunnel. This number has been chosen to be on the safe side, since the airfoils with reflexed trailing edge are particularly sensitive at low Reynold numbers when it comes to creation of unwanted bubbles and low pressure wakes in the boundary layer, that are the results of small disturbances.
To specify the critical amplification exponent, we first enter the boundary layer parameters menu:

> **VPAR**

Then we enter the value of the exponent:

> **NCRIT "9"**

To exit the **VPAR** menu and go back to the OPER menu, we press return one time.
When we now have specified the conditions, we need to tell XFOIL where and how to save our analysis. To do this we type the command for auto polar accumulation, which is:

> PACC

XFOIL will now request a filename, which we want to save as a .dat-file. It is simply done by ending the file name with that suffix. A good idea can be to give the file a name that specifies the conditions that the test was made at. This makes it easy to know what every file contains, under what conditions, and what airfoil that was used in the analysis. Here is an example:

> Enter polar save filename OR <return> for no file s> naca4412_re400000.dat

After pressing return, we are ready to do the analysis. The Reynold number and Mach number is now fixed, and cannot be changed midstream (when a polar accumulation has already started). XFOIL will calculate and output the basic data for the conditions once the angle of attack OR the lift coefficient is specified and prescribed. These can be typed in at a specific fixed value or as a sequence of values within a span. For a specific value on angle of attack, the following command shall be typed:

> ALFA "number" (in degrees)

For a specific lift coefficient:

> CL "number"

To do a sequence of values on the angle of attack, the span and number of calculations in this span needs to be specified. XFOIL will ask for the starting value and ending value on the angle of attack for the sequence, as well as the increment between every calculation. These are asked for when the command for the angle of attack sequence is typed, which is as follows:

> ASEQ

or simply

> AS

XFOIL will display:

> Enter first alfa value (deg) r> "number"
> Enter last alfa value (deg) r> "number"
> Enter alfa increment (deg) r> "number"

We can either type that way, or all the numbers directly after ASEQ, separated with a space. For example, if the desired sequence for testing is between an angle of attack of -2 degrees up to 13 degrees, with an increment of 1 degree per calculation, we type the following:

> AS "-2 13 1"

Respectively, the command for a sequence with the lift coefficient as variable, the command is as follows:

> CSEQ

or simply

> CS

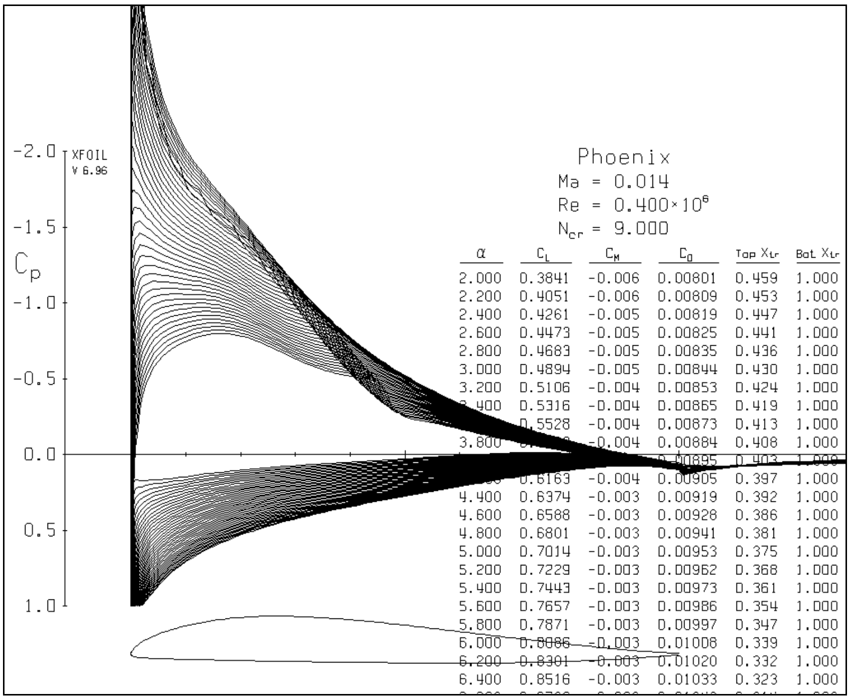

Figure 8. Example of how XFOIL displays a diagram of graphs, with a sequence of alphas where the calculations were successful.

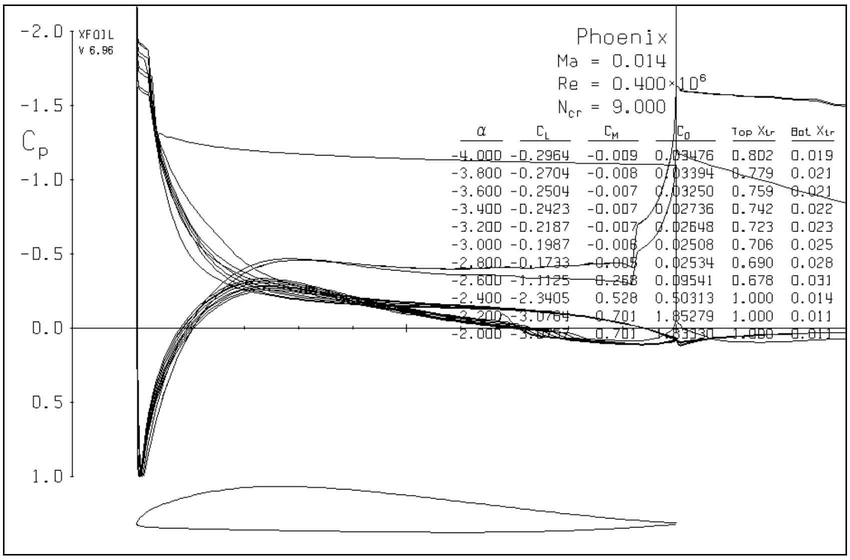

Figure 9. Example of how XFOIL displays a diagram of graphs, with a sequence of alphas where XFOIL fails to solve all the calculations. Many of the graphs look jagged and are quite irregular. The sequence is incomplete. XFOIL was only able to give the solution for -4 to -2 degrees in angle of attack, when the intention actually was to reach up to 10 degrees.

Once the sequence is specified, XFOIL will perform the calculations. If no problems occur, we simply type the PACC command once again, to save the data to the chosen file.
Problems often occur but can be solved by modification and smoothing to the panel and its nodes. Some airfoils and areas are more sensitive then others, and it can therefore be good to do the sequence in smaller steps, and lower the increment number. If the calculations repeatedly fail at some or many of the steps when doing a sequence and XFOIL, it can be a good idea to force a re-initialization of the boundary layer. This shall be done at the step in the sequence that was just before the step where the program failed. To do this at the angle of attack where the last successful calculation was done, we use the following commands to perform the task. First we enter the menu for the boundary layer parameters:

> **VPAR**

Then we use the command to force the re-initialization:

> **INIT**

When a successful run has been saved, to display the C_L – C_D graphs we simply type the command to load the saved polar file:

> **PGET "filename"**

The polars have now been loaded into XFOIL, and to display them as a graph we need to type the command for that:

> **PPLO**

Example of XFOIL plot

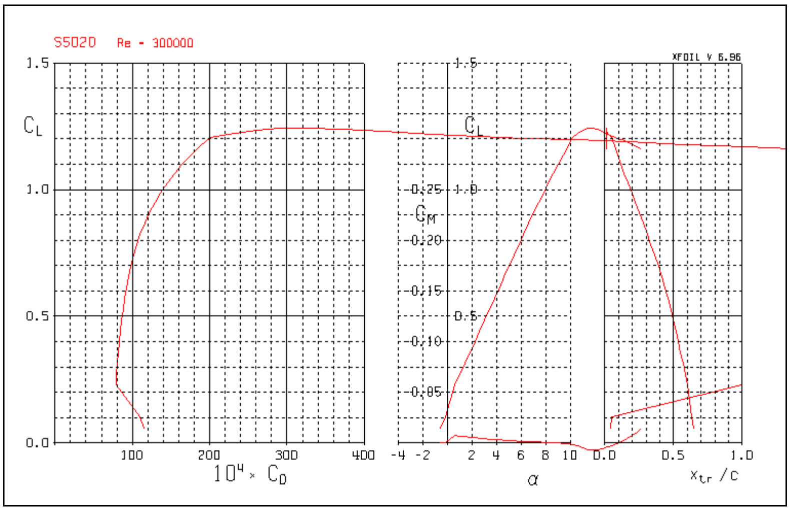

Figure 10. Example of how a graph is displayed inside XFOIL after using the PPLO command.

All the files that have been saved after typing the PACC command a second time, are stored as .txt files in the main folder of XFOIL. In these files, the raw data of every sequence is divided into rows and columns, where the columns are separated only by a space. To calculate the glide ratio as well as the endurance, the numbers need to be copied from the text document, and then pasted into Excel. In Excel the columns and rows can be divided into cells. Eventually, the calculations of both endurance and glide ratio can be realized and displayed very simply as colourful tables, thanks to the many utilities of Excel.

```
       XFOIL         Version 6.96

 Calculated polar for: S5020

 1 1 Reynolds number fixed          Mach number fixed

 xtrf =   1.000 (top)        1.000 (bottom)
 Mach =   0.000     Re =     0.400 e 6     Ncrit =   9.000

  alpha    CL        CD       CDp       CM     Top_Xtr  Bot_Xtr
 ------ -------- --------- --------- -------- -------- --------
 -0.200   0.1003   0.01008   0.00237   0.0010   0.6087   0.0371
  1.000   0.2696   0.00700   0.00180  -0.0063   0.5623   1.0000
  1.400   0.3123   0.00710   0.00176  -0.0058   0.5467   1.0000
  1.800   0.3552   0.00720   0.00175  -0.0052   0.5301   1.0000
  2.200   0.3980   0.00734   0.00177  -0.0047   0.5136   1.0000
  2.600   0.4409   0.00747   0.00182  -0.0043   0.4956   1.0000
  3.000   0.4839   0.00763   0.00191  -0.0039   0.4770   1.0000
  3.400   0.5268   0.00782   0.00201  -0.0035   0.4580   1.0000
  3.800   0.5697   0.00802   0.00216  -0.0031   0.4372   1.0000
  4.200   0.6126   0.00825   0.00234  -0.0028   0.4152   1.0000
  4.600   0.6553   0.00852   0.00255  -0.0025   0.3915   1.0000
  5.000   0.6980   0.00882   0.00281  -0.0022   0.3662   1.0000
  5.400   0.7405   0.00917   0.00312  -0.0020   0.3397   1.0000
  5.800   0.7827   0.00959   0.00347  -0.0017   0.3128   1.0000
  6.200   0.8248   0.01004   0.00389  -0.0015   0.2849   1.0000
  6.600   0.8665   0.01058   0.00437  -0.0013   0.2575   1.0000
  7.000   0.9079   0.01113   0.00491  -0.0012   0.2299   1.0000
  7.400   0.9488   0.01179   0.00551  -0.0010   0.2035   1.0000
  7.800   0.9892   0.01248   0.00619  -0.0008   0.1758   1.0000
  8.200   1.0286   0.01330   0.00693  -0.0006   0.1482   1.0000
  8.600   1.0677   0.01413   0.00773  -0.0004   0.1221   1.0000
  9.000   1.1058   0.01506   0.00865  -0.0001   0.0994   1.0000
  9.400   1.1427   0.01609   0.00966   0.0002   0.0788   1.0000
  9.800   1.1776   0.01732   0.01081   0.0006   0.0558   1.0000
 10.200   1.2056   0.01941   0.01269   0.0014   0.0212   1.0000
 10.600   1.2296   0.02175   0.01509   0.0027   0.0117   1.0000
 11.000   1.2505   0.02403   0.01757   0.0040   0.0101   1.0000
 11.400   1.2653   0.02643   0.02019   0.0055   0.0091   1.0000
 11.800   1.2650   0.02955   0.02350   0.0072   0.0088   1.0000
 12.200   1.2640   0.03354   0.02768   0.0072   0.0083   1.0000
 12.600   1.2522   0.03927   0.03360   0.0062   0.0080   1.0000
 13.000   1.2434   0.04506   0.03959   0.0047   0.0077   1.0000
 13.400   1.2363   0.05090   0.04562   0.0029   0.0076   1.0000
 13.800   1.2268   0.05728   0.05220   0.0008   0.0075   1.0000
 14.200   1.2158   0.06412   0.05923  -0.0017   0.0073   1.0000
 14.600   1.2055   0.07116   0.06646  -0.0044   0.0073   1.0000
 15.000   1.1954   0.07858   0.07407  -0.0074   0.0072   1.0000
 15.400   1.1846   0.08651   0.08219  -0.0109   0.0072   1.0000
 15.800   1.1725   0.09498   0.09088  -0.0148   0.0071   1.0000
```

Figure 11. A text document containing the raw-data from a sequence in XFOIL.

5.6.7 Forced transition

Some airfoils need turbulators. These can be simulated by forcing the transition from laminar to turbulent at any chosen spot on the airfoil. It's done in the menu for the boundary layer parameters (**VPAR**), which in turn is found inside the analysis routine menu (**OPER**).
Once inside the VPAR menu, the command for forcing the transition is as follows:

> XTR

XFOIL will ask for the locations both on the top side and the bottom side of the airfoil. This is done in percent of the chord, counting from the leading edge of the airfoil. It looks as follows:

> Enter top side Xtrip/c r> "number"
> Enter bottom side Xtrip/c r> "number"

If only one side is desired, we simply press return on the other, and no forced transition will be performed on that side.

5.7 Calculating the Reynold numbers

Before starting the calculations in XFOIL, what needs to be done is finding the Reynold numbers for Solaris.

We have chosen three different guesstimated airspeeds at which the aircraft will operate in: 5, 10, and 15m/s. What we first need to do is to calculate the Reynold numbers for the specific airspeeds.

The Reynold number is being calculated by using the simple formula:

$$\mathbf{Re} = \boldsymbol{\rho}\ \mathbf{V}\ \mathbf{c}\ /\ \boldsymbol{\mu} \quad (7)$$

where **ρ** is the density of the air, **V** is the free stream airspeed, **c** is the chord length, and **μ** is the absolute dynamic viscosity coefficient.

Our chord length is 0,29m.

For standard sea level:

T = 288,16 K
p = 101325 N/m^2
ρ = 1,225 kg/m^3
μ = 1,7894 • 10^{-5}kg/ms

The Reynold numbers are displayed in the table below:

The Reynold numbers

m/s	Re
Re_{cr}	50.000
Re_{cr}	100.000
5	200.000
10	300.000
15	400.000

Table 1. The table shows the five Reynold numbers that the airfoils will be tested at.

5.8 Turbulators

All airfoils at low Reynold numbers are in the danger zone of having laminar separation due to bubbles in the boundary layer. This is caused by pressure rising along the airfoil creating a much thicker boundary layer than wanted. A thick boundary layer is extremely sensitive to turbulence. In optimal conditions it might work flying with, but only very little disturbance in the flow is enough to make the layer fail, creating a wake with uncontrolled and turbulent flow. This wake is called a laminar separation bubble, which of course creates a massive amount of drag. The phenomenon can be avoided by forcing the airflow from laminar to turbulent by adding so called turbulators.
Turbulent but controlled airflow is preferred over uncontrolled wakes that build up behind the separation point, since these have a much lower pressure coefficient than rest the boundary layer which as already said, results in high amounts of additional drag. The turbulent flow has a very high amount of kinetic energy, which forces it to follow the shape of the airfoil. The disadvantage of having turbulent flow is the increased amount of skin friction drag. Comparing the two choices clearly shows that forcing the flow from laminar to turbulent gives the best advantages. The rise in skin friction drag is way less than the amount a drag created by a separation bubble.
Reflexed airfoils are extra sensitive to these bubbles. The reflexed shape and trailing edge increases the pressure along the top surface of the airfoil and encourages laminar separation even more.
By forcing a transition point (the point where the airflow goes from laminar to turbulent) in XFOIL we can simulate such a turbulator and create a controlled turbulent flow on the wing. Placing the transition point close to the leading edge would create a massive amount of unnecessary drag (due to increased friction between the turbulent air and airfoil surface) along the whole airfoil. Placing the transition point too far back would risk bubbles forming before the turbulators.

5.9 Early tests

Two of the chosen airfoils (HS 3.0/9.0B and HS 3.0/8.0B) seem to have the tendency for creating bubbles along the bottom side of the airfoil. Through trial and error it seems like adding turbulators at about 70% of the chord line (counting from the leading edge) on the bottom side of the airfoil gives the best results. Moving them forward towards the leading edge means playing safer, but it adds massive amounts of drag and has negative effects on the lift coefficient (it means that we are forcing the air way too early and that we are flying in more turbulent air than is actually needed). Having the turbulators further back (lets say on 80%) makes them very sensitive to separation and starts having a tendency to creating bubbles.

6 Results

6.1 Endurance table for all airfoils

This table shows the maximum endurance the airfoils that were calculated in XFOIL. This was done at the three chosen guesstimated Reynold numbers as well as with the two lower, critical Reynold numbers. All candidate airfoils were included in this analysis, and are now displayed in the table below.
The three airfoils marked with red color were disqualified in the very early stage of the analysis and is therefore not presented with any data. It was quickly shown that these airfoils were way too thick for such low Reynold numbers that Solaris is designed for. The six airfoils marked with green color were the best ones, qualifying for the next elimination round.

Endurance table

Airfoil	***Re=50.000***	***Re=100.000***	***Re=200.000***	***Re=300.000***	***Re=400.000***
Phoenix	25,52	49,47	67,74	78,26	85,64
HS 3.0/9.0B	15,46	48,48	65,01	73,34	79,06
S5020	25,47	45,14	61,98	71,98	78,83
S5010	22,10	44,23	60,66	70,37	77,20
EH 2.0/10	15,75	46,82	65,40	70,67	76,23
HS 3.0/8.0B	30,33	49,12	63,21	70,17	75,59
HS 520	26,23	43,10	58,60	67,74	73,91
HS 2.0/8.0	29,43	44,34	58,63	71,89	73,55
HS 522	27,29	42,59	57,57	66,48	72,74
MH 60	26,24	42,46	57,26	66,29	72,66
SD7003-085-88	25,90	38,77	52,81	63,43	70,59
Sipkill 1.7/10B	28,81	43,37	57,03	64,36	69,63
MH 45	26,30	41,19	55,39	63,63	68,82
HS 130	26,42	40,75	54,87	63,08	68,80
MH 61	28,65	43,63	56,80	63,92	68,51
MH 62	26,88	42,13	54,65	62,84	68,41
MH 44	27,14	40,60	53,77	62,22	68,03
MH 46	27,91	41,35	54,36	61,38	66,22
EH 1.5/9.0	28,42	41,98	53,95	60,20	64,56
MH 64	26,77	39,15	51,84	58,89	64,08
~~MH 49~~					
~~LA2573A~~					
~~HS 3.4/12B~~					

Table 2. The table displays the endurance of all the tested airfoils. The airfoils are ranked after the maximum value of endurance on Re = 400.000.

6.2 Data on a higher scale – comparison of the best airfoils

Reflexed airfoils are due to their complex shape, a lot more difficult to work with in XFOIL, compared to other ordinary airfoils. The sensitiveness of reflexed airfoils is more significant in the lower range of Reynold numbers. Problems often occur around zero angle of attack (**α**, which is the same as *alpha* in the text), and down to the negative values, as well as close to where the lift coefficient (**C_L**) reaches its maximum. However, negative values on alpha are not in the interest of this project, since the relevant envelope only includes positive alphas.
When observing the diagrams below, a few drag-curves look better than other. With the optimal drag-curve in mind, it is easy to see these wanted characteristics.
What we want is the steep drag curve up in the left areas of the diagram. Both the S5010 and the S5020 have a very good shape on this curve, where the S5020 is slightly better than the S5010. The width, of a good value on endurance is large on the airfoil. However, comparing the airfoil with the one that has the best peak of endurance, which is the Phoenix, the later one is the favoured one. Let's observe the alphas and the values close to where maximum endurance is found. At higher alphas, the Phoenix has a larger drop than the S5020, which explains the shape of the curves in the graphs of Phoenix. However, at lower values of alpha, close to the peak of endurance, the Phoenix is still higher than the S5020. It is more likely to operate the aircraft at slightly lower alphas than at slightly higher alphas, of where the maximum endurance is found. At very low alphas the S5020 is slightly better, but with a very close margin. However, where the guesstimated operating of the aircraft is going to be, which is at alphas close to and below the peak of endurance, the Phoenix has a greater value on this important parameter.

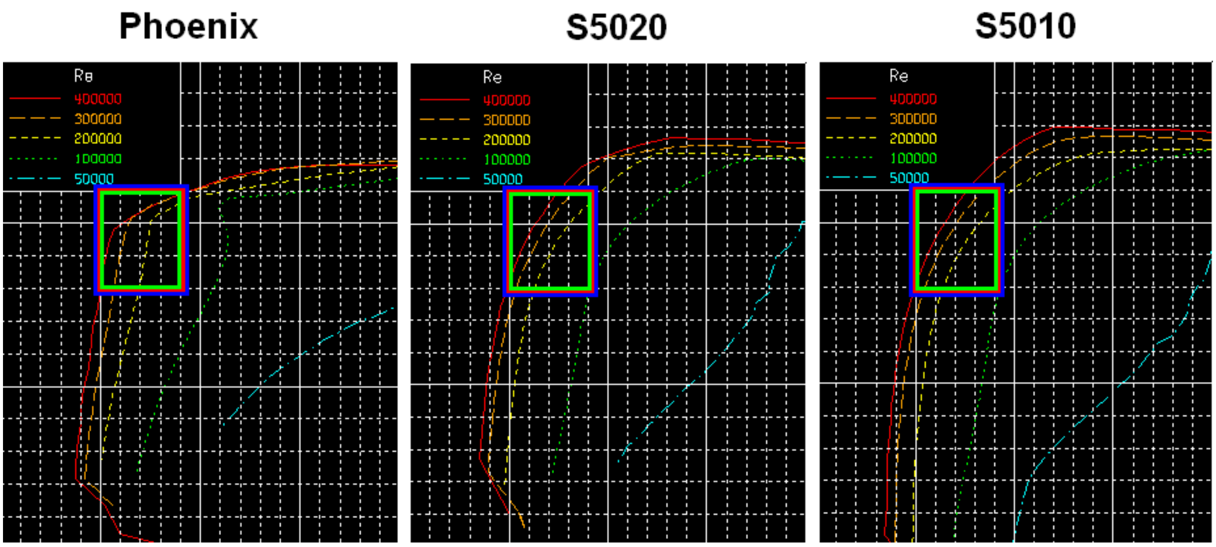

Figure 12. C_L vs C_D diagrams for the top three airfoils. By looking in the interesting area where maximum endurance is found (inside the multi-coloured square) it is clear that Phoenix is closer to the left, and because of that has a higher peak than the other two airfoils.

7 Winner airfoil – summary and conclusions

The task was to determine a suitable airfoil for Solaris. This was done by comparing different airfoils in XFOIL, to eventually find the one with the best endurance characteristics. Before testing the airfoils, a study was made to gather a pool of potential airfoils. The method during the search, when the goal was to find suiting airfoils for the project, was to look for reflexed trailing edge airfoils that work good on low Reynold numbers. This was done by making a field study on airfoil data bank websites, as well as discussing different airfoils and their characteristics on various internet forums. Once the pool of airfoils was gathered, they were all tested at different Reynold numbers inside XFOIL. By exporting the raw data given by the program, and processing it inside Excel, the airfoils could be ranked in terms of endurance. The ranking shows that Phoenix is the airfoil superior to all the other tested low Reynold number reflexed airfoil. Phoenix was designed by Brett von Perlick/Kowalski and was first used on a radio controlled plank flying wing called Phönix, which is why the airfoil has gotten the English translation Phoenix. The airfoil has a significantly higher value on endurance, in the range of alphas where the craft is planned to operate in. This is true close and below the point of where maximum endurance is found. At higher alphas, the Phoenix has a lower value than the second best airfoil, which is the S 5020.

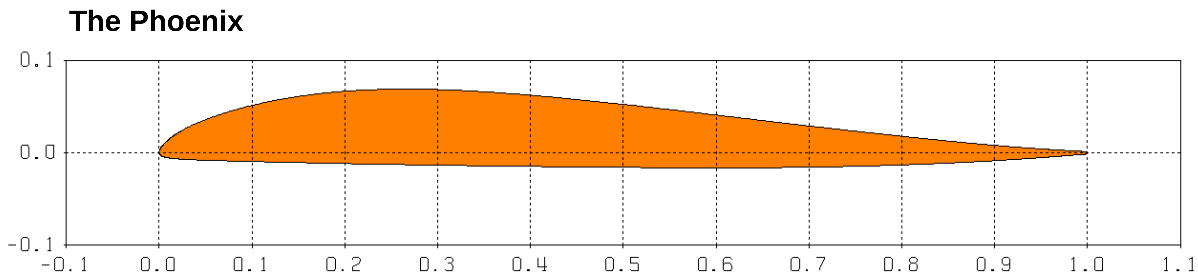

Figure 13. The geometric shape of the winner airfoil, the Phoenix.

8 Calculating the lift force

Lift force is calculated by the simple formula:

$$\mathbf{L = q \cdot S \cdot C_L} \tag{5}$$

For the thought Reynold numbers at which the vehicle is planned to operate in (Re = 400.000 – 200.000), CL varies from 0,980 to 1,017 (at the angle of attack where we find the best endurance). A mean value of these will be used in the calculations:

$$\mathbf{C_L} = 1$$

The dynamic pressure varies with speed:

$$\mathbf{q = ½ \cdot \rho \cdot v^2} \tag{6}$$

The density of air for standard sea level is:

$$\boldsymbol{\rho} = 1{,}225 kg/m^3$$

Symbol explanation:
q is the dynamic pressure
ρ is the air density
S is the wing area
L is the lifting force
C_L is the lift coefficient

Lift force

v = 5m/s
q = 15,31Pa

	Area (m^2)	Lift (N)	Lift (kg)
3-panel version	0,87	13,32	1,36
4-panel version	1,16	17,76	1,81
5-panel version	1,45	22,20	2,26

v = 10m/s
q = 61,25Pa

	Area (m^2)	Lift (N)	Lift (kg)
3-panel version	0,87	53,29	5,43
4-panel version	1,16	71,05	7,24
5-panel version	1,45	88,81	9,04

v = 15m/s
q = 112,50Pa

	Area (m^2)	Lift (N)	Lift (kg)
3-panel version	0,87	97,88	9,97
4-panel version	1,16	130,50	13,29
5-panel version	1,45	163,13	16,61

Table 3. Displaying the lift force of Phoenix.

9 Calculating the hinge moments

The hinge moments are being tested and evaluated on the highest Reynold number and with the highest guesstimated speed of the vehicle. This is done because of the actuators being the same on all three versions of Solaris. It demands that the actuators manage the most extreme cases of hinge moment. These cases are found on the configuration of Solaris that give the highest values on the parameter. The hinge moments that are presented are tested when the airfoil is flying when the angle of attack of the airfoil is 8 degrees. This is the alfa at which the Phoenix will be cruising at.

α = 8°
Re = 400.000
V = 15m/s
M = 0,0441 (at standard sea level)

	Flap deflection (degrees)	$C_{hinge\ moment}$
	-20	0,001289
	-15	0,000957
Up	-10	0,000572
	-5	0,000211
	0	0,000698
	5	0,000576
	10	0,001097
	15	0,001542
	20	0,001900
Down	25	0,002311
	30	0,002746
	35	0,003062
	40	0,003447
	45	0,003243
	50	0,003891

Table 4. Flap sized to 10% of cord length

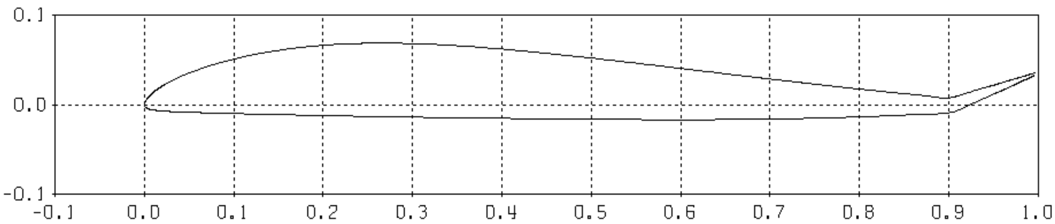

Figure 14. The 10% flap version of Phoenix at -20 degrees of flap deflection.

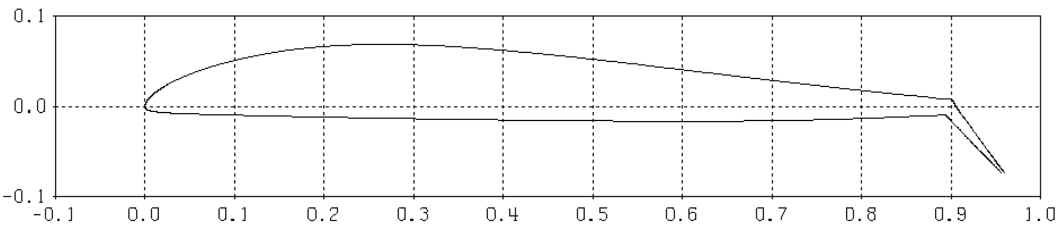

Figure 15. The 10% flap version of Phoenix at 50 degrees of flap deflection.

	Flap deflection (degrees)	$C_{hinge\ moment}$
	-20	0,010586
	-15	0,007063
Up	-10	0,006971
	-5	0,000851
	0	0,000698
	5	0,002800
	10	0,004386
	15	0,006185
	20	0,008059
Down	25	0,009673
	30	0,011495
	35	0,011487
	40	0,014780
	45	0,013541
	50	0,018409

Table 5. Flap sized to 20% of chord length.

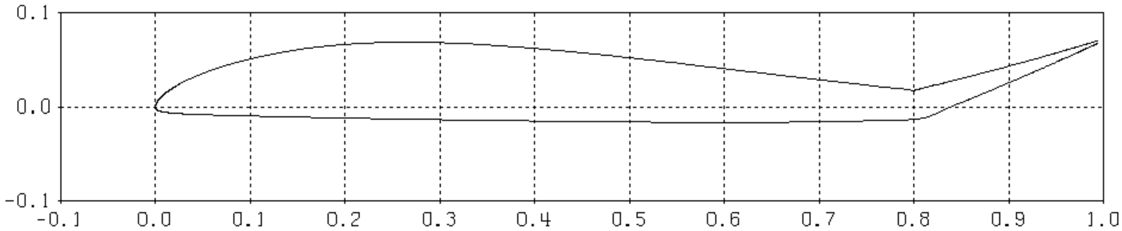

Figure 16. The 20% flap version of Phoenix at -20 degrees of flap deflection

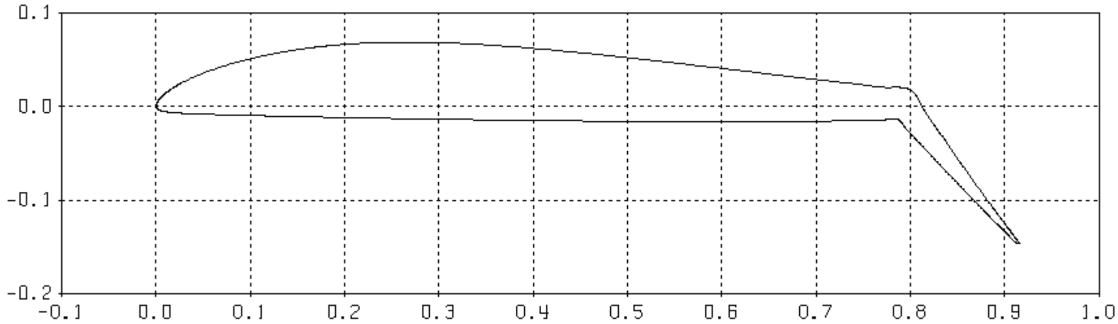

Figure 17. The 20% flap version of Phoenix at 50 degrees of flap deflection

10 Comparison airfoil – Eppler 387

The popular Eppler 387 was used as a comparison airfoil because of its common use in the same range of Reynold numbers as what Solaris will operate in. It has been widely used on radio controlled soaring planes. The airfoil has a very good glide ratio, which goes hand in hand with endurance.

The comparison shows the lower values of Phoenix, towards the higher values of Eppler 387. When comparing the lift coefficient values, the difference is not that remarkable. However, when looking at the endurance and glide ratio, the drop in performance is quite extensive. The quotes are higher on the Phoenix, because of the significantly higher drag coefficient on the airfoil. This is true for the values close to where maximum endurance is found. The results show how difficult reflexed airfoils are to use at very low Reynold numbers. The lower the Reynold number, the worse the reflexed airfoil behaves compared to the ordinary airfoil. The difference between the two airfoils is most likely less at higher Reynold numbers than the tested ones.
However, by comparing the airfoils and looking in the tables, in the range of where the both airfoils have their maximum lift coefficient, the endurance is higher for the Phoenix. The increase in drag is significant for Phoenix, and clearly has to do with reflexed airfoils and their behaviour at low Reynold numbers. An increase of the freestream speed will increase the Reynold number, and lower the difference.

Comparing two different aircrafts, one with a conventional airfoil (designed with a fuselage and a tail) and one with a reflexed trailing edge airfoil (plank flying wing), the both designs clearly have different negative effects and positive advantages.
The conventional airfoil needs to have a fuselage and a tail, who both have very negative effects on the aircraft in different ways. A fuselage needs to be dimensioned to carry the tail, which requires a certain amount of structural strength and material. This results in extra weight. The same goes for the tail, which is required in the design, when using a conventional airfoil like the Eppler 387. Both tail and fuselage also comes with extra skin friction drag, which in turn requires more thrust from the engines to maintain level flight..
Drag needs to be minimized, when the aim is to maximize endurance. Thanks to Solaris simple design, the negative effects from a fuselage and a tail can be neglected.

Comparison of maximum values

		Eppler 387	Phoenix	Difference in % (How much lower Phoenix is than the Eppler 387)
Re = 400.000	(C_L/C_D)	108,33	86,73	-19,9%
	$(C_L^{3/2})/C_D$	109,74	85,64	-22,0%
	C_L	1,3676	1,1811	-13,6%
Re = 300.000	(C_L/C_D)	96,31	78,13	-18,9%
	$(C_L^{3/2})/C_D$	101,29	78,26	-22,7%
	C_L	1,3551	1,198	-11,6%
Re = 200.000	(C_L/C_D)	84,23	67,16	-20,3%
	$(C_L^{3/2})/C_D$	89,05	67,74	-23,9%
	C_L	1,3329	1,186	-11,0%
Re = 100.000	(C_L/C_D)	60,95	48,23	-20,9%
	$(C_L^{3/2})/C_D$	66,05	49,47	-25,1%
	C_L	1,3221	1,1883	-10,1%
Re = 50.000	(C_L/C_D)	38,29	24,32	-36,5%
	$(C_L^{3/2})/C_D$	41,82	25,52	-39,0%
	C_L	1,2931	1,1432	-11,6%

Table 6. Comparision of maximum values of the Phoenix and the Eppler 387.

11 References

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Last updated (Wednesday, 22 April, 2009)
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http://web.mit.edu/drela/Public/web/xfoil

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The official website of XFOIL. Version used for this project: XFOIL v6.96.
Last visited: 27-07-2011

http://yahoo.com/groups/xfoil

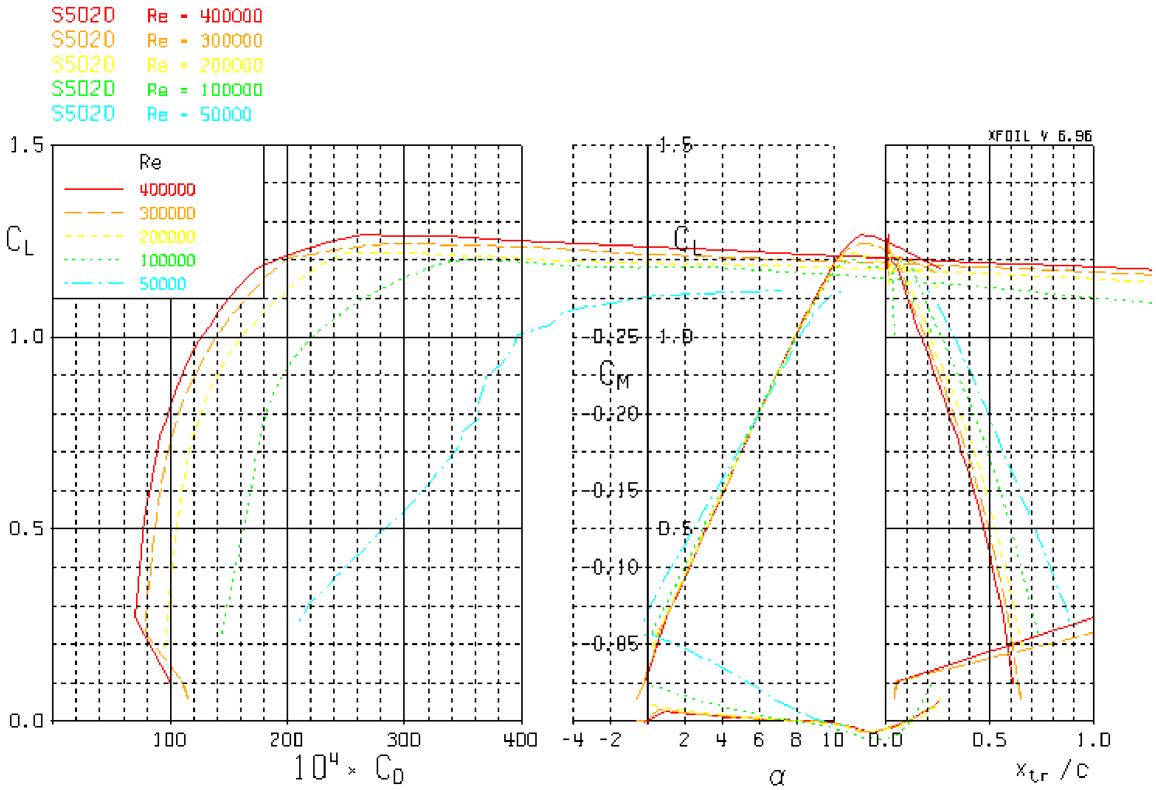
XFOIL discussion forum, with the creator of the program (Mark Drela) as participant.

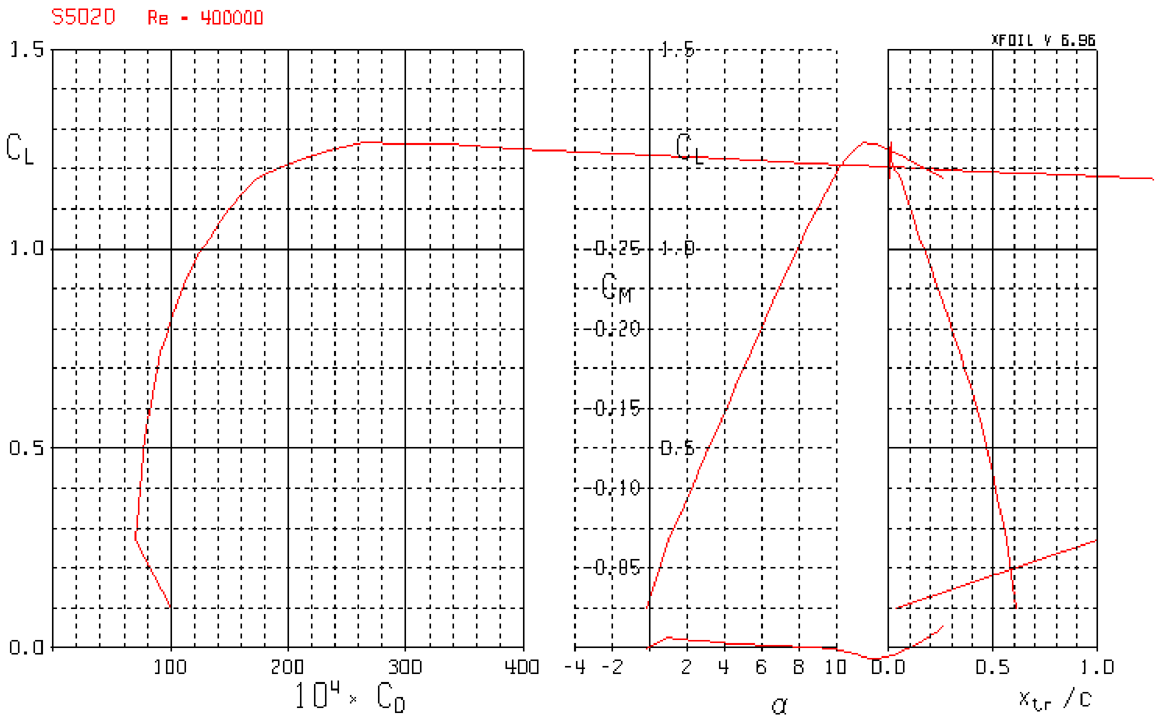
12 Appendix

Color representation for the marked values in the tables

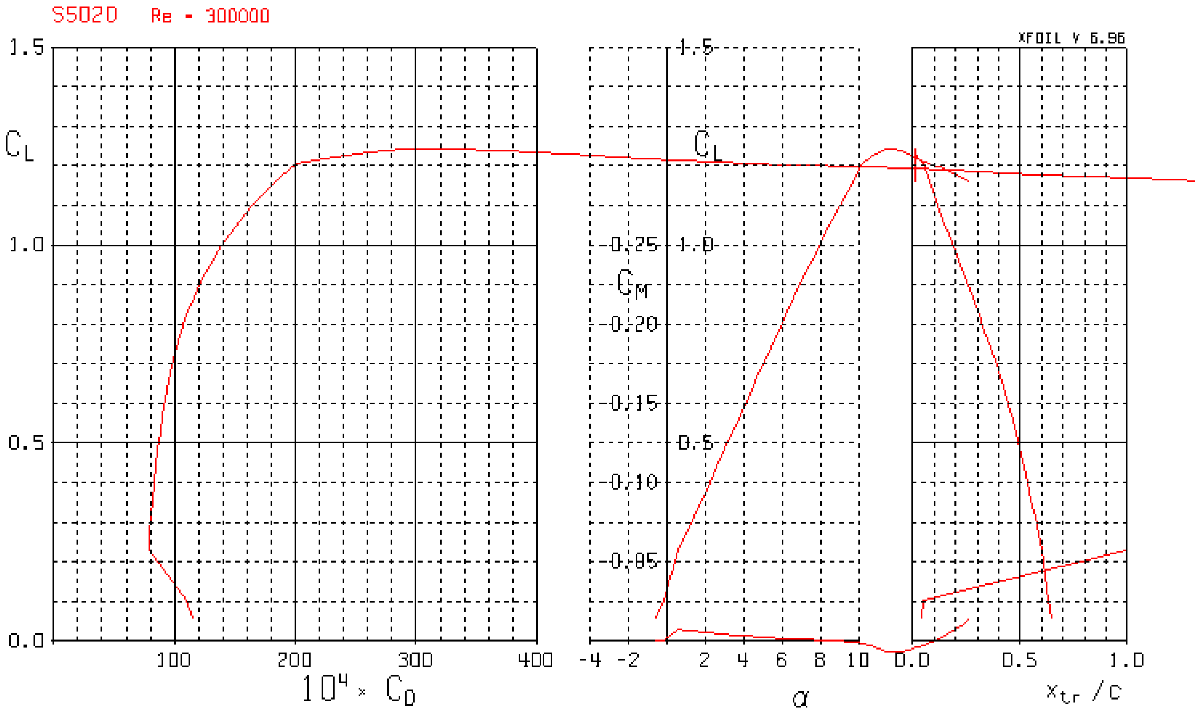
- (cyan) $(L/D)_{max}$
- (green) $(L^{3/2}/D)_{max}$
- (orange) C_{Lmax}

12.1.1 **Airfoil ID: *S5020***

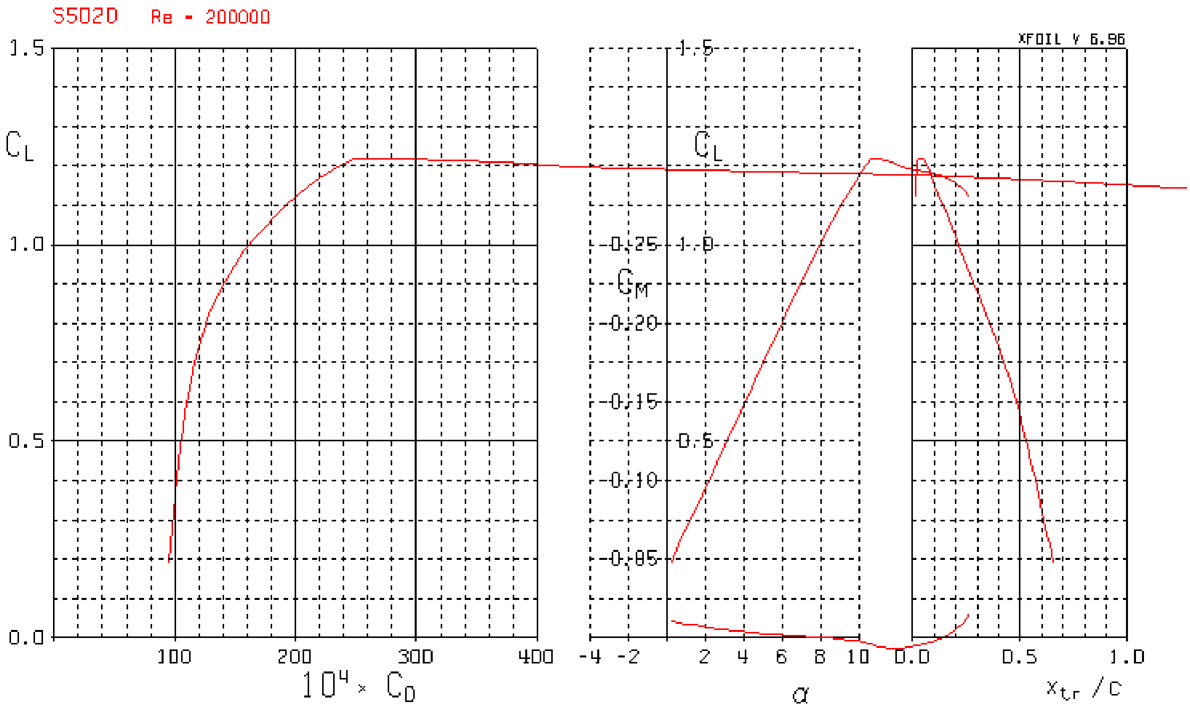




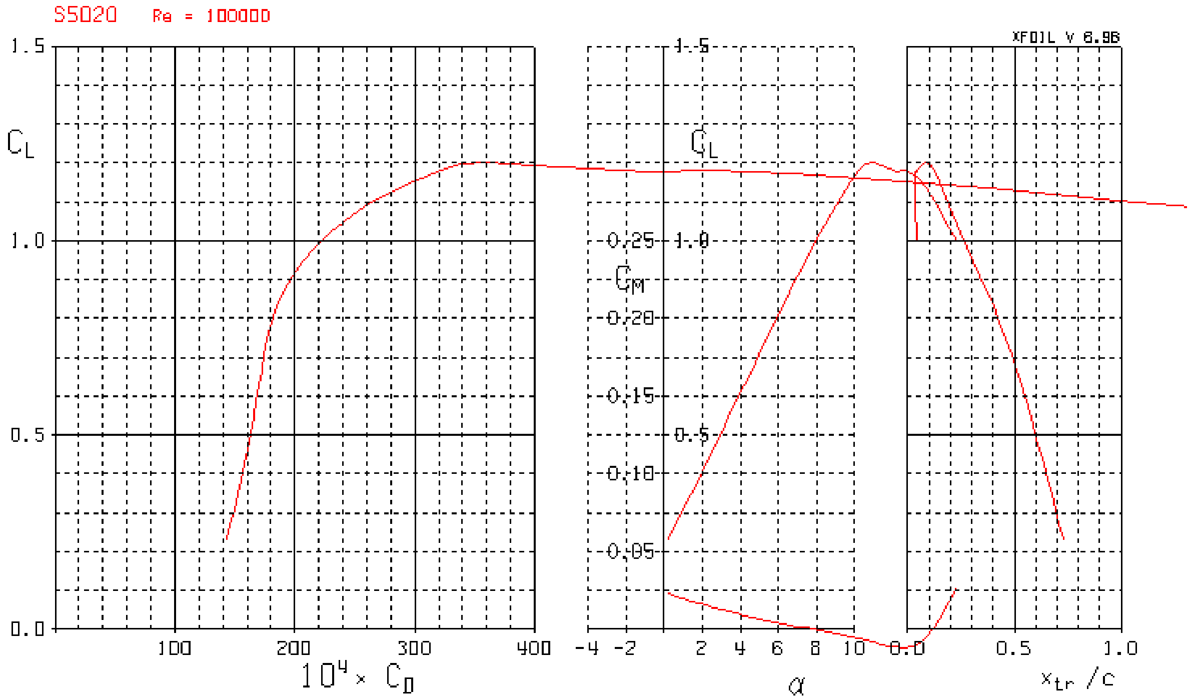
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
9,95	-0,2	0,1003	0,0101	3,15
38,51	1,0	0,2696	0,0070	20,00
43,99	1,4	0,3123	0,0071	24,58
49,33	1,8	0,3552	0,0072	29,40
54,22	2,2	0,3980	0,0073	34,21
59,02	2,6	0,4409	0,0075	39,19
63,42	3,0	0,4839	0,0076	44,12
67,37	3,4	0,5268	0,0078	48,89
71,03	3,8	0,5697	0,0080	53,62
74,25	4,2	0,6126	0,0083	58,12
76,91	4,6	0,6553	0,0085	62,26
79,14	5,0	0,6980	0,0088	66,12
80,75	5,4	0,7405	0,0092	69,49
81,62	5,8	0,7827	0,0096	72,21
82,15	6,2	0,8248	0,0100	74,61
81,90	6,6	0,8665	0,0106	76,24
81,57	7,0	0,9079	0,0111	77,73
80,47	7,4	0,9488	0,0118	78,39
79,26	7,8	0,9892	0,0125	78,83
77,34	8,2	1,0286	0,0133	78,44
75,56	8,6	1,0677	0,0141	78,08
73,43	9,0	1,1058	0,0151	77,21
71,02	9,4	1,1427	0,0161	75,92
67,99	9,8	1,1776	0,0173	73,78
62,11	10,2	1,2056	0,0194	68,20
56,53	10,6	1,2296	0,0218	62,69
52,04	11,0	1,2505	0,0240	58,19
47,87	11,4	1,2653	0,0264	53,85
42,81	11,8	1,2650	0,0296	48,15
37,69	12,2	1,2640	0,0335	42,37
31,89	12,6	1,2522	0,0393	35,68
27,59	13,0	1,2434	0,0451	30,77
24,29	13,4	1,2363	0,0509	27,01
21,42	13,8	1,2268	0,0573	23,72
18,96	14,2	1,2158	0,0641	20,91
16,94	14,6	1,2055	0,0712	18,60
15,21	15,0	1,1954	0,0786	16,63
13,69	15,4	1,1846	0,0865	14,90
12,34	15,8	1,1725	0,0950	13,37



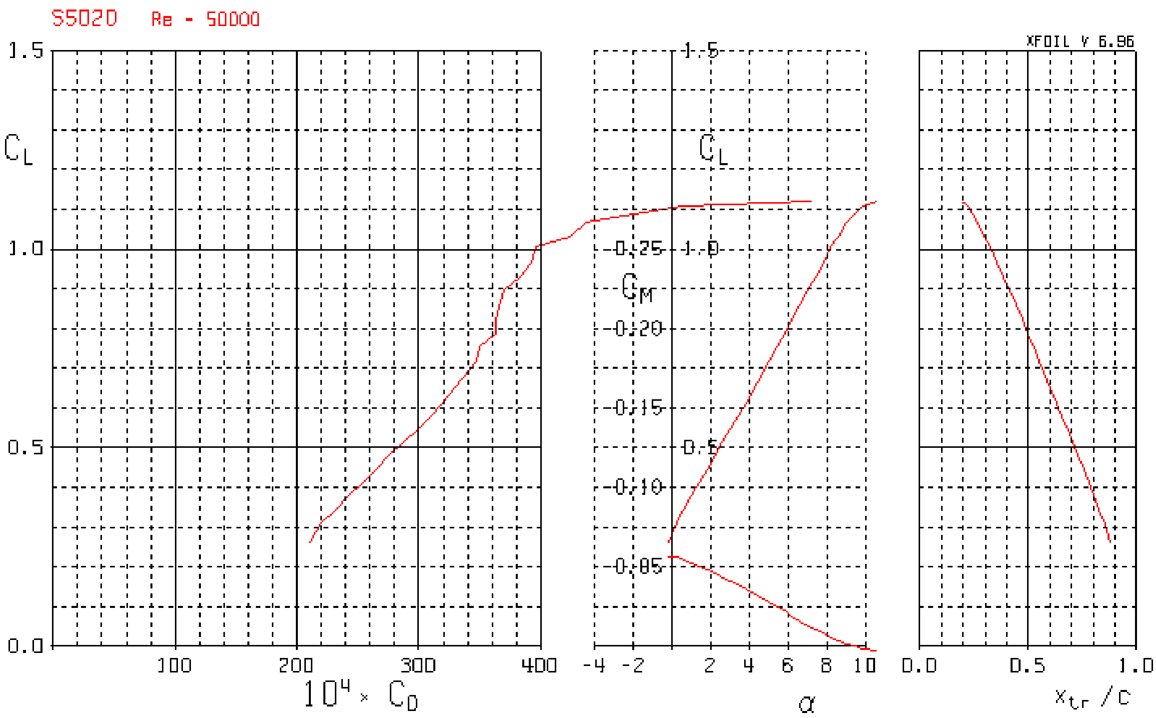
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
5,19	-0,6	0,0600	0,0116	1,27
9,42	-0,2	0,1036	0,0110	3,03
5,19	-0,6	0,0600	0,0116	1,27
9,42	-0,2	0,1036	0,0110	3,03
5,19	-0,6	0,0600	0,0116	1,27
9,42	-0,2	0,1036	0,0110	3,03
29,19	0,6	0,2294	0,0079	13,98
34,15	1,0	0,2718	0,0080	17,80
39,00	1,4	0,3143	0,0081	21,86
43,70	1,8	0,3570	0,0082	26,11
48,09	2,2	0,3996	0,0083	30,40
52,34	2,6	0,4423	0,0085	34,81
56,34	3,0	0,4851	0,0086	39,24
59,98	3,4	0,5278	0,0088	43,57
63,46	3,8	0,5705	0,0090	47,93
66,52	4,2	0,6133	0,0092	52,09
69,11	4,6	0,6559	0,0095	55,97
71,35	5,0	0,6985	0,0098	59,63
73,14	5,4	0,7409	0,0101	62,95
74,14	5,8	0,7829	0,0106	65,60
74,85	6,2	0,8249	0,0110	67,99
74,75	6,6	0,8663	0,0116	69,57
74,44	7,0	0,9074	0,0122	70,91
73,53	7,4	0,9478	0,0129	71,59
72,42	7,8	0,9878	0,0136	71,98
70,76	8,2	1,0267	0,0145	71,70
69,12	8,6	1,0651	0,0154	71,33
66,95	9,0	1,1020	0,0165	70,28
64,90	9,4	1,1383	0,0175	69,24
62,62	9,8	1,1729	0,0187	67,82
59,90	10,2	1,2051	0,0201	65,75
52,42	10,6	1,2218	0,0233	57,94
47,35	11,0	1,2362	0,0261	52,64
42,83	11,4	1,2413	0,0290	47,72
38,04	11,8	1,2409	0,0326	42,38
33,40	12,2	1,2384	0,0371	37,17
28,96	12,6	1,2307	0,0425	32,12
24,98	13,0	1,2183	0,0488	27,57
22,08	13,4	1,2097	0,0548	24,29
19,81	13,8	1,2039	0,0608	21,74
17,77	14,2	1,1959	0,0673	19,44
16,03	14,6	1,1878	0,0741	17,47
14,49	15,0	1,1790	0,0814	15,73
13,08	15,4	1,1684	0,0893	14,14
11,77	15,8	1,1554	0,0982	12,65



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
20,37	0,2	0,1941	0,0095	8,97
24,48	0,6	0,2360	0,0096	11,89
28,49	1,0	0,2778	0,0098	15,02
32,33	1,4	0,3197	0,0099	18,28
36,12	1,8	0,3619	0,0100	21,73
39,71	2,2	0,4039	0,0102	25,24
43,28	2,6	0,4462	0,0103	28,91
46,62	3,0	0,4886	0,0105	32,59
49,75	3,4	0,5308	0,0107	36,24
52,73	3,8	0,5732	0,0109	39,92
55,51	4,2	0,6156	0,0111	43,55
57,96	4,6	0,6579	0,0114	47,02
60,09	5,0	0,7001	0,0117	50,28
61,95	5,4	0,7422	0,0120	53,37
63,22	5,8	0,7839	0,0124	55,97
64,13	6,2	0,8254	0,0129	58,27
64,36	6,6	0,8663	0,0135	59,90
64,17	7,0	0,9067	0,0141	61,10
63,61	7,4	0,9465	0,0149	61,88
62,44	7,8	0,9853	0,0158	61,98
61,13	8,2	1,0233	0,0167	61,84
57,56	9,0	1,0953	0,0190	60,24
55,68	9,4	1,1292	0,0203	59,17
53,69	9,8	1,1613	0,0216	57,86
51,62	10,2	1,1914	0,0231	56,34
48,87	10,6	1,2173	0,0249	53,92
42,22	11,0	1,2197	0,0289	46,63
32,51	11,8	1,2092	0,0372	35,75
28,05	12,2	1,1985	0,0427	30,71
24,97	12,6	1,1950	0,0479	27,30
22,39	13,0	1,1910	0,0532	24,43
20,24	13,4	1,1873	0,0587	22,05
18,32	13,8	1,1825	0,0645	19,93
16,59	14,2	1,1757	0,0709	17,99
15,00	14,6	1,1674	0,0778	16,21
13,48	15,0	1,1559	0,0858	14,49
12,00	15,4	1,1402	0,0950	12,81
10,57	15,8	1,1194	0,1059	11,18

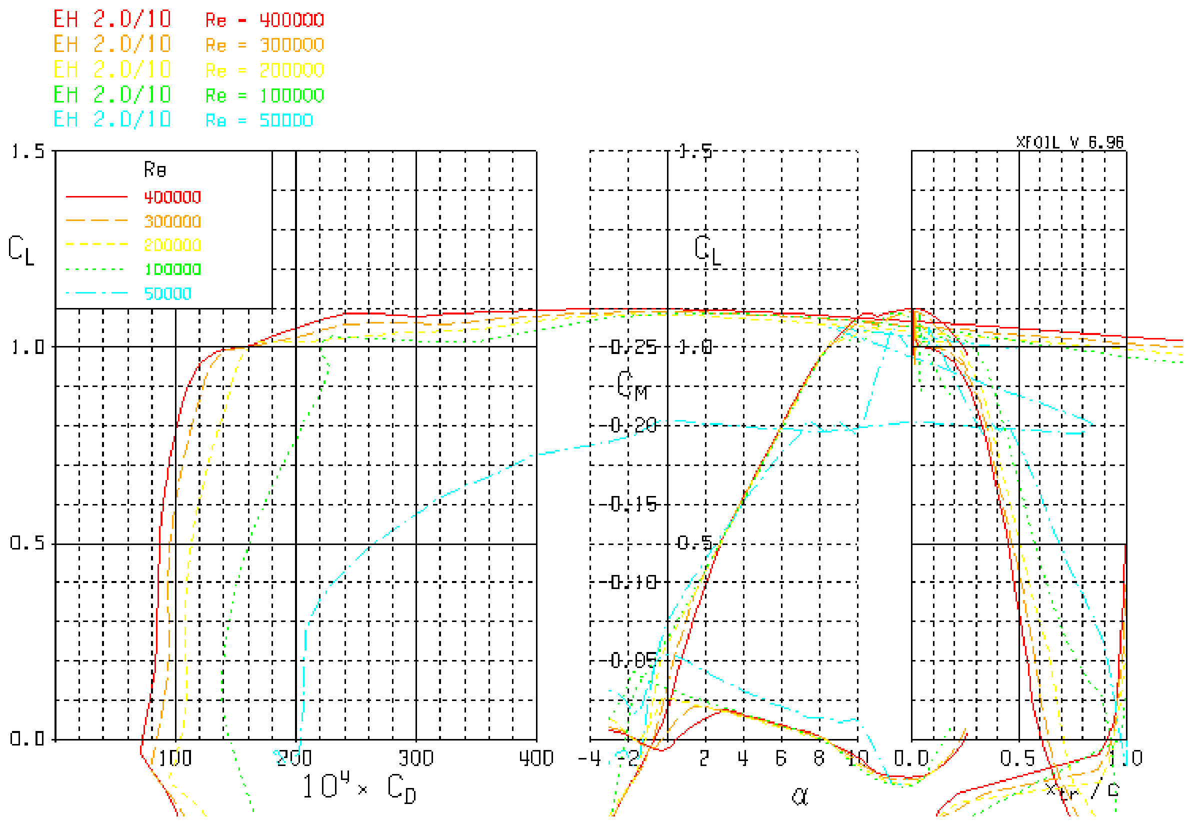


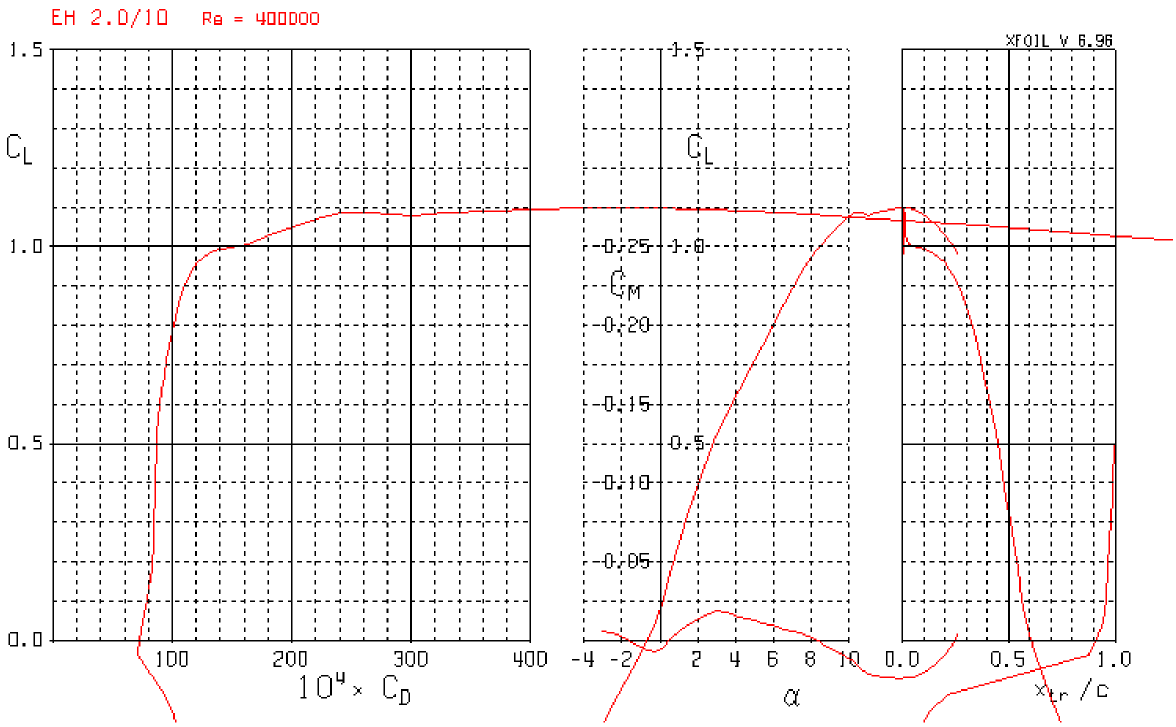
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
16,10	0,2	0,2311	0,0144	7,74
18,42	0,6	0,2697	0,0146	9,57
20,64	1,0	0,3085	0,0150	11,46
22,82	1,4	0,3468	0,0152	13,44
24,93	1,8	0,3866	0,0155	15,50
26,99	2,2	0,4261	0,0158	17,62
29,09	2,6	0,4655	0,0160	19,85
31,07	3,0	0,5058	0,0163	22,10
33,04	3,4	0,5461	0,0165	24,41
34,99	3,8	0,5865	0,0168	26,80
36,93	4,2	0,6271	0,0170	29,25
38,80	4,6	0,6677	0,0172	31,70
40,57	5,0	0,7084	0,0175	34,15
42,27	5,4	0,7490	0,0177	36,58
43,57	5,8	0,7895	0,0181	38,71
44,85	6,2	0,8298	0,0185	40,86
45,58	6,6	0,8696	0,0191	42,50
45,95	7,0	0,9088	0,0198	43,80
45,90	7,4	0,9473	0,0206	44,67
45,47	7,8	0,9849	0,0217	45,13
44,66	8,2	1,0214	0,0229	45,14
42,08	9,0	1,0904	0,0259	43,95
38,51	9,8	1,1527	0,0299	41,35
36,73	10,2	1,1805	0,0321	39,91
35,38	10,6	1,1960	0,0338	38,70
33,30	11,0	1,2028	0,0361	36,52
30,10	11,4	1,1958	0,0397	32,91
26,57	11,8	1,1839	0,0446	28,92
23,75	12,2	1,1793	0,0497	25,79
21,64	12,6	1,1833	0,0547	23,54
19,11	13,0	1,1724	0,0613	20,70
16,80	13,4	1,1571	0,0689	18,07
14,58	13,8	1,1342	0,0778	15,52
12,45	14,2	1,1031	0,0886	13,07
10,52	14,6	1,0675	0,1014	10,87
8,89	15,0	1,0313	0,1160	9,03
7,61	15,4	0,9990	0,1314	7,60



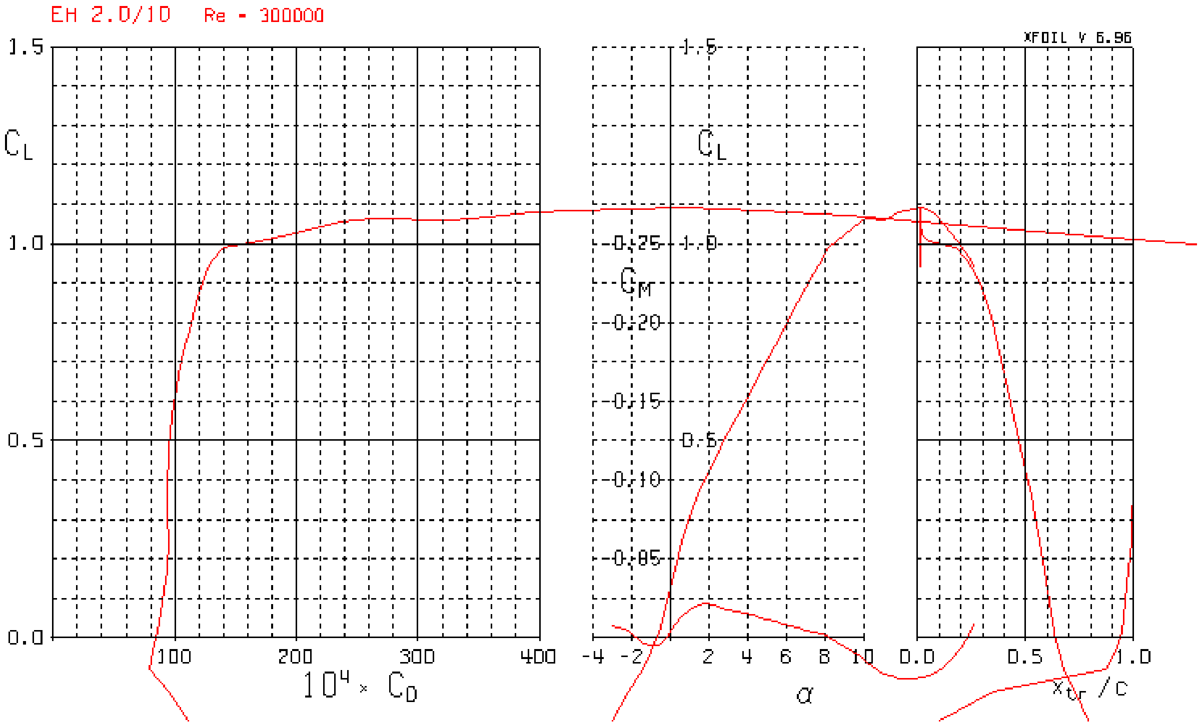
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
12,53	-0,2	0,2636	0,0210	6,43
14,03	0,2	0,3093	0,0220	7,80
14,86	0,6	0,3441	0,0232	8,72
15,60	1,0	0,3788	0,0243	9,60
16,26	1,4	0,4130	0,0254	10,45
16,87	1,8	0,4468	0,0265	11,27
17,44	2,2	0,4804	0,0276	12,09
17,83	2,6	0,5127	0,0288	12,76
18,21	3,0	0,5454	0,0300	13,45
18,67	3,4	0,5790	0,0310	14,21
19,21	3,8	0,6133	0,0319	15,04
20,00	4,6	0,6805	0,0340	16,50
20,63	5,0	0,7157	0,0347	17,45
21,54	5,4	0,7527	0,0350	18,68
21,65	5,8	0,7854	0,0363	19,19
22,69	6,2	0,8236	0,0363	20,60
24,26	7,0	0,8978	0,0370	22,99
24,27	7,4	0,9309	0,0384	23,42
24,67	7,8	0,9668	0,0392	24,26
25,40	8,2	1,0058	0,0396	25,47
24,31	8,6	1,0320	0,0425	24,69
24,44	9,0	1,0686	0,0437	25,26
22,81	9,4	1,0881	0,0477	23,79
21,47	9,8	1,1075	0,0516	22,60
20,47	10,2	1,1293	0,0552	21,76
13,91	10,6	1,0248	0,0737	14,08

12.1.2 **Airfoil ID: *EH 2.0 / 10.0***

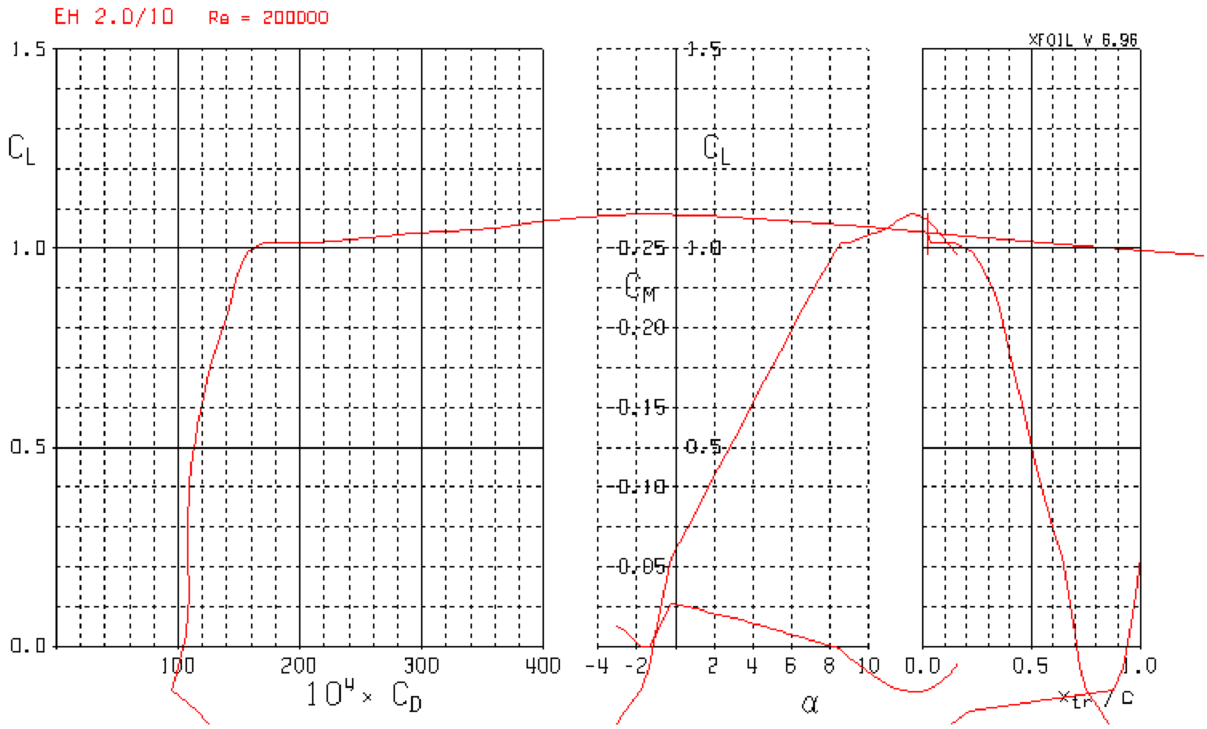




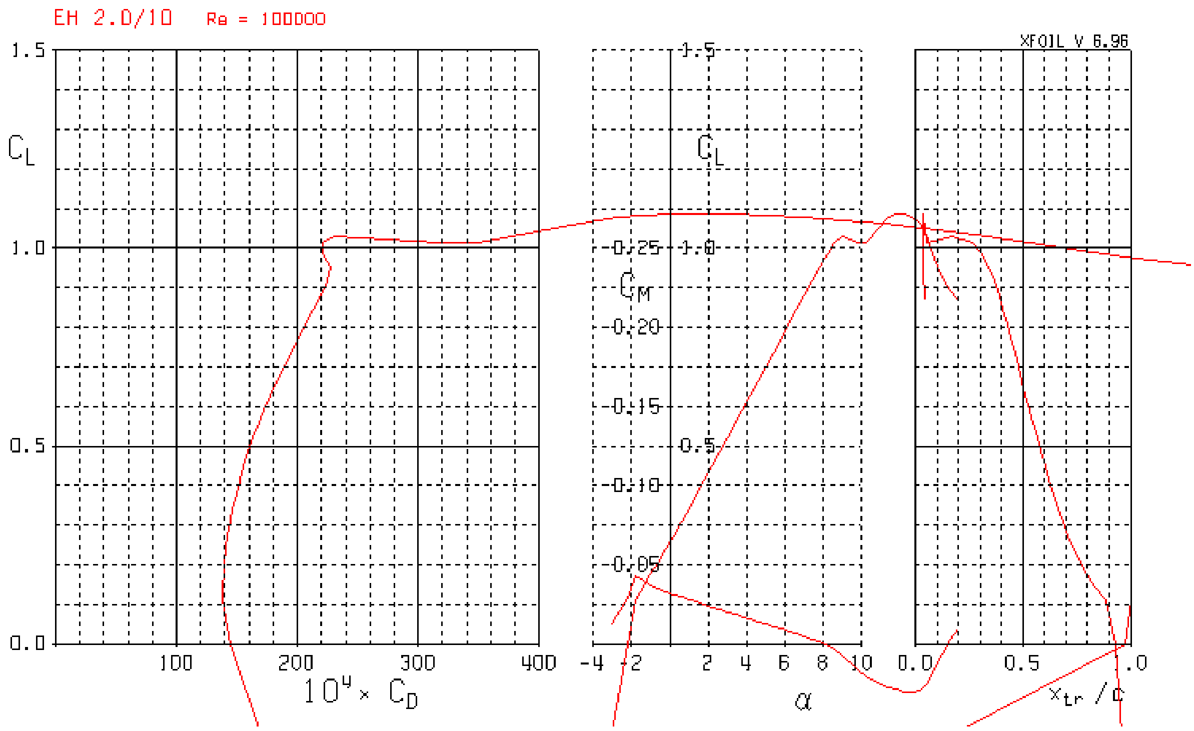
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,18	-0,6	0,0013	0,0073	0,01
6,07	-0,2	0,0458	0,0075	1,30
13,30	0,2	0,1045	0,0079	4,30
22,30	0,6	0,1835	0,0082	9,55
29,50	1,0	0,2481	0,0084	14,69
36,04	1,4	0,3056	0,0085	19,92
42,24	1,8	0,3607	0,0085	25,37
48,58	2,2	0,4178	0,0086	31,40
54,81	2,6	0,4741	0,0087	37,74
60,22	3,0	0,5245	0,0087	43,61
63,78	3,4	0,5619	0,0088	47,81
67,19	3,8	0,5993	0,0089	52,01
70,08	4,2	0,6363	0,0091	55,90
72,45	4,6	0,6731	0,0093	59,44
74,89	5,0	0,7100	0,0095	63,11
76,89	5,4	0,7466	0,0097	66,44
78,30	5,8	0,7830	0,0100	69,29
79,90	6,2	0,8198	0,0103	72,35
80,99	6,6	0,8561	0,0106	74,94
81,49	7,0	0,8923	0,0110	76,98
81,11	7,4	0,9279	0,0114	78,13
79,45	7,8	0,9622	0,0121	77,94
73,91	8,2	0,9911	0,0134	73,58
61,98	8,6	1,0065	0,0162	62,18
56,98	9,0	1,0307	0,0181	57,84
48,58	9,8	1,0702	0,0220	50,26
45,18	10,2	1,0848	0,0240	47,06
41,18	10,6	1,0855	0,0264	42,90
35,86	11,0	1,0772	0,0300	37,22
33,04	11,4	1,0860	0,0329	34,43
29,77	11,8	1,0898	0,0366	31,08
27,16	12,2	1,0956	0,0403	28,43
24,63	12,6	1,0987	0,0446	25,82
22,28	13,0	1,0993	0,0493	23,36
19,98	13,4	1,0957	0,0548	20,91
17,73	13,8	1,0867	0,0613	18,49
15,53	14,2	1,0712	0,0690	16,08
13,58	14,6	1,0525	0,0775	13,93
11,74	15,0	1,0292	0,0876	11,91
10,21	15,4	1,0060	0,0985	10,24
8,83	15,8	0,9795	0,1109	8,74



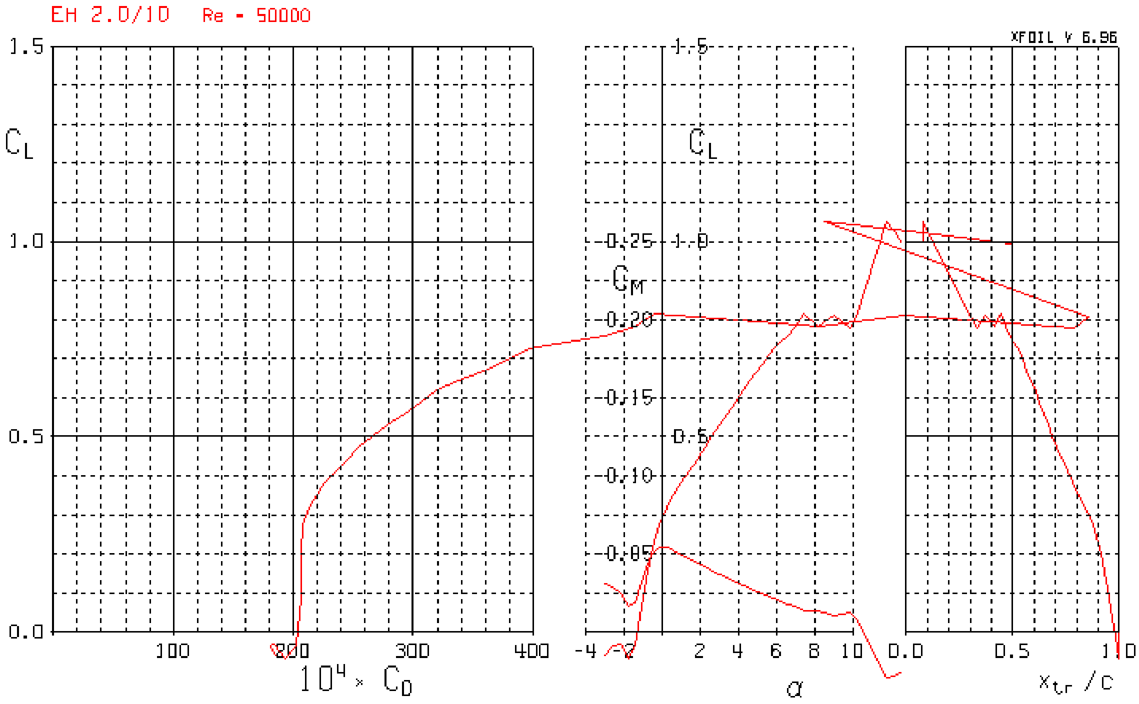
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
1,57	-0,60	0,0134	0,0085	0,18
10,10	-0,2	0,0908	0,0090	3,04
18,51	0,2	0,1734	0,0094	7,71
25,91	0,6	0,2456	0,0095	12,84
32,39	1,0	0,3051	0,0094	17,89
38,57	1,4	0,3622	0,0094	23,21
43,60	1,8	0,4094	0,0094	27,90
47,19	2,2	0,4464	0,0095	31,53
50,75	2,6	0,4836	0,0095	35,29
53,97	3,0	0,5208	0,0097	38,95
57,13	3,4	0,5582	0,0098	42,69
59,95	3,8	0,5953	0,0099	46,25
62,33	4,2	0,6320	0,0101	49,55
64,63	4,6	0,6689	0,0104	52,86
66,63	5,0	0,7056	0,0106	55,97
68,22	5,4	0,7422	0,0109	58,77
69,72	5,8	0,7788	0,0112	61,53
71,04	6,2	0,8155	0,0115	64,15
72,26	6,6	0,8520	0,0118	66,70
73,34	7,0	0,8881	0,0121	69,11
73,86	7,4	0,9240	0,0125	71,00
73,42	7,8	0,9589	0,0131	71,90
70,89	8,2	0,9910	0,0140	70,57
59,57	8,6	1,0056	0,0169	59,74
52,60	9,0	1,0230	0,0195	53,20
44,46	9,8	1,0564	0,0238	45,70
40,57	10,2	1,0617	0,0262	41,80
32,39	11,0	1,0597	0,0327	33,34
29,87	11,4	1,0696	0,0358	30,89
27,41	11,8	1,0790	0,0394	28,47
25,31	12,2	1,0854	0,0429	26,37
23,15	12,6	1,0899	0,0471	24,16
20,98	13,0	1,0902	0,0520	21,90
18,67	13,4	1,0837	0,0581	19,43
16,38	13,8	1,0697	0,0653	16,94
14,08	14,2	1,0465	0,0744	14,40
12,19	14,6	1,0231	0,0839	12,33
10,39	15,0	0,9937	0,0956	10,36
8,95	15,4	0,9663	0,1080	8,80
7,66	15,8	0,9361	0,1222	7,41



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
2,94	-1,0	0,0313	0,0106	0,52
11,90	-0,6	0,1307	0,0110	4,30
20,90	-0,2	0,2251	0,0108	9,92
24,28	0,2	0,2608	0,0107	12,40
27,61	0,6	0,2968	0,0108	15,04
30,88	1,0	0,3332	0,0108	17,83
34,05	1,4	0,3698	0,0109	20,71
37,18	1,8	0,4068	0,0109	23,72
40,11	2,2	0,4440	0,0111	26,73
42,96	2,6	0,4812	0,0112	29,80
45,53	3,0	0,5181	0,0114	32,77
47,94	3,4	0,5551	0,0116	35,71
50,08	3,8	0,5920	0,0118	38,54
52,01	4,2	0,6288	0,0121	41,24
53,80	4,6	0,6655	0,0124	43,89
55,34	5,0	0,7023	0,0127	46,38
56,72	5,4	0,7390	0,0130	48,76
57,94	5,8	0,7758	0,0134	51,03
58,97	6,2	0,8126	0,0138	53,16
59,97	6,6	0,8492	0,0142	55,27
60,96	7,0	0,8851	0,0145	57,35
62,23	7,4	0,9197	0,0148	59,68
63,16	7,8	0,9543	0,0151	61,70
63,16	8,2	0,9879	0,0156	62,78
59,62	8,6	1,0147	0,0170	60,05
47,71	9,0	1,0157	0,0213	48,08
38,37	9,8	1,0317	0,0269	38,97
35,60	10,2	1,0382	0,0292	36,28
32,45	10,6	1,0408	0,0321	33,11
29,07	11,0	1,0483	0,0361	29,76
27,04	11,4	1,0657	0,0394	27,92
24,95	11,8	1,0782	0,0432	25,91
22,68	12,2	1,0846	0,0478	23,62
20,29	12,6	1,0809	0,0533	21,10
18,04	13,0	1,0691	0,0593	18,65
15,84	13,4	1,0531	0,0665	16,25
13,83	13,8	1,0318	0,0746	14,04
11,96	14,2	1,0077	0,0842	12,01
10,41	14,6	0,9806	0,0942	10,31

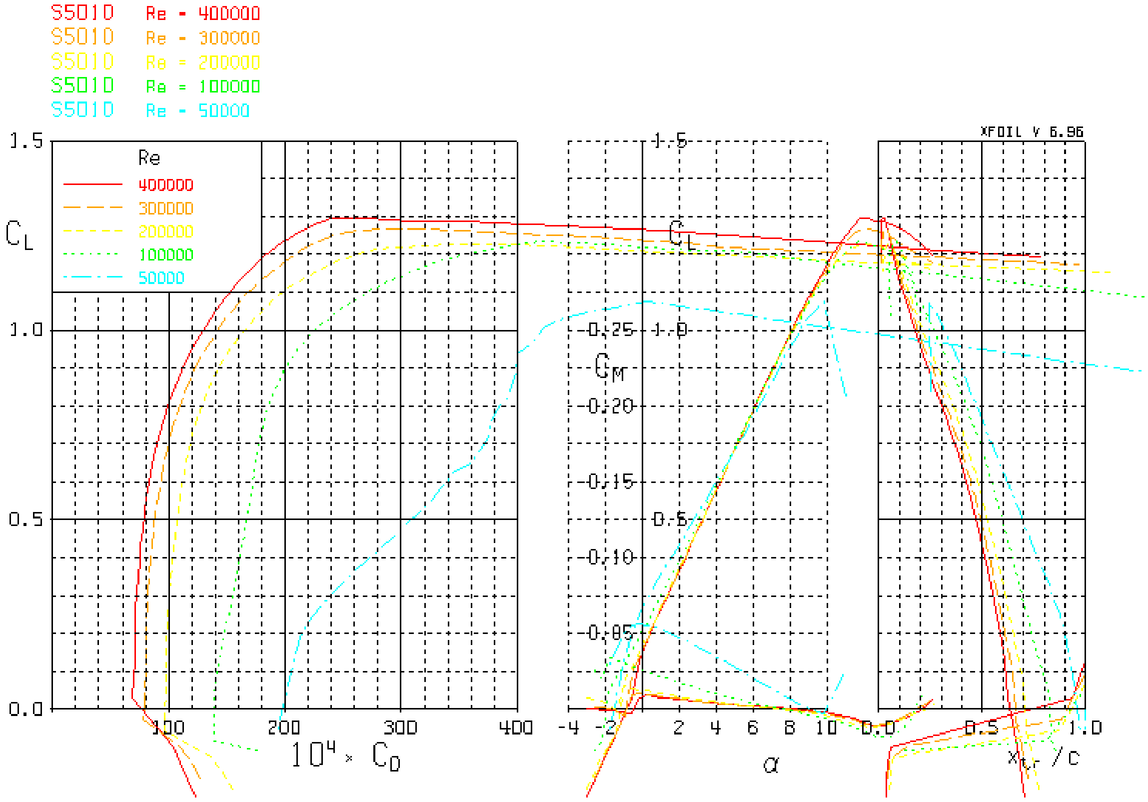


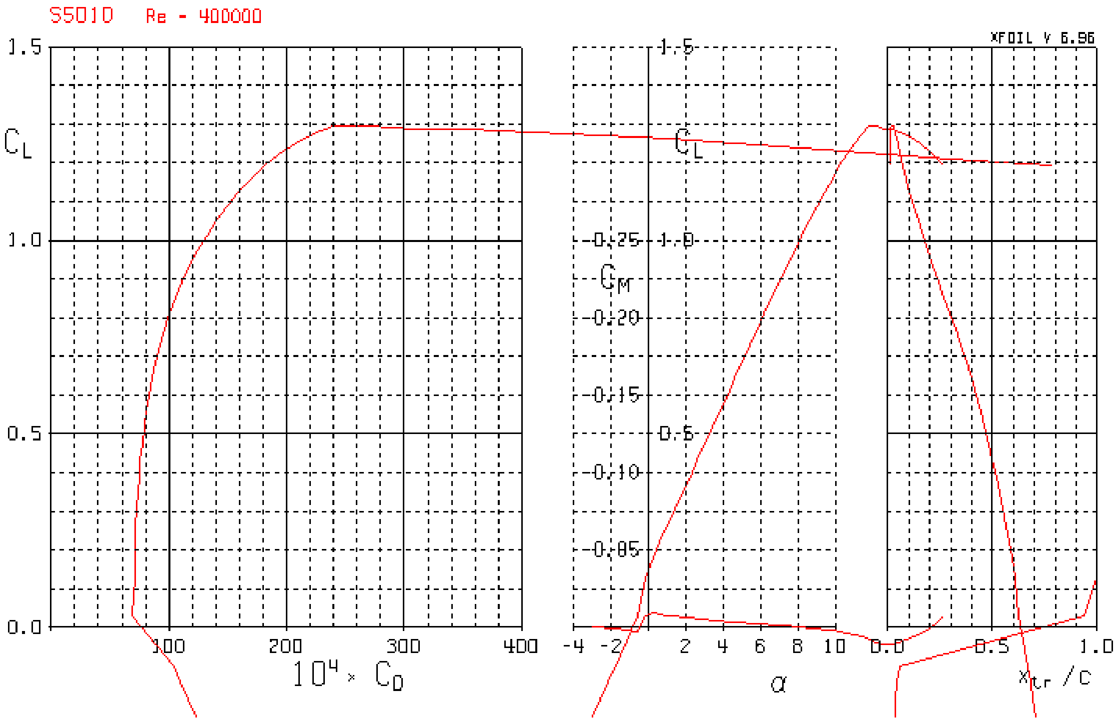
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
8,17	-1,8	0,1127	0,0138	2,74
10,41	-1,4	0,1438	0,0138	3,95
12,51	-1,0	0,1737	0,0139	5,22
14,71	-0,6	0,2056	0,0140	6,67
16,92	-0,2	0,2387	0,0141	8,27
19,12	0,2	0,2726	0,0143	9,98
21,28	0,6	0,3073	0,0144	11,80
23,36	1,0	0,3425	0,0147	13,67
25,30	1,4	0,3778	0,0149	15,55
27,14	1,8	0,4134	0,0152	17,45
28,84	2,2	0,4491	0,0156	19,33
30,43	2,6	0,4848	0,0159	21,19
31,83	3,0	0,5205	0,0164	22,97
33,14	3,4	0,5565	0,0168	24,73
34,32	3,8	0,5924	0,0173	26,42
35,34	4,2	0,6283	0,0178	28,01
36,32	4,6	0,6646	0,0183	29,61
37,06	5,0	0,7005	0,0189	31,02
37,90	5,4	0,7371	0,0195	32,54
38,41	5,8	0,7728	0,0201	33,77
39,22	6,2	0,8102	0,0207	35,30
39,57	6,6	0,8453	0,0214	36,38
40,06	7,0	0,8813	0,0220	37,61
40,79	7,4	0,9177	0,0225	39,07
41,87	7,8	0,9526	0,0228	40,87
44,49	8,2	0,9846	0,0221	44,15
45,91	8,6	1,0128	0,0221	46,20
44,70	9,0	1,0290	0,0230	45,34
31,88	9,8	1,0123	0,0318	32,08
29,01	10,2	1,0132	0,0349	29,20
25,10	11,0	1,0537	0,0420	25,77
23,21	11,4	1,0774	0,0464	24,10
21,36	11,8	1,0870	0,0509	22,27
19,45	12,2	1,0871	0,0559	20,28
17,46	12,6	1,0790	0,0618	18,14
15,51	13,0	1,0612	0,0684	15,97
13,59	13,4	1,0332	0,0761	13,81
10,88	13,8	0,9730	0,0894	10,74
8,93	14,2	0,9264	0,1037	8,60
7,53	14,6	0,8911	0,1183	7,11
6,61	15,0	0,8692	0,1314	6,17



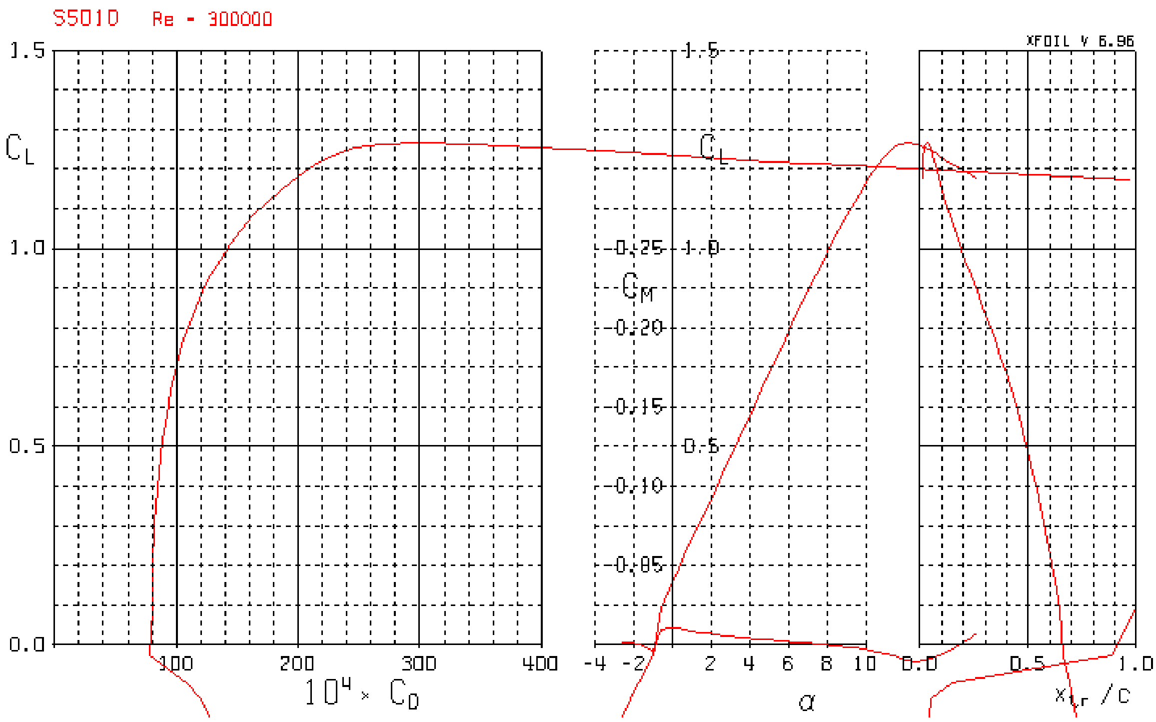
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
4,20	-1,0	0,0868	0,0207	1,24
9,71	-0,6	0,2007	0,0207	4,35
13,12	-0,2	0,2729	0,0208	6,85
14,93	0,2	0,3172	0,0213	8,41
16,02	0,6	0,3512	0,0219	9,49
16,95	1,0	0,3838	0,0226	10,50
17,55	1,4	0,4132	0,0235	11,28
18,10	1,8	0,4429	0,0245	12,05
18,54	2,2	0,4724	0,0255	12,74
18,86	2,6	0,5015	0,0266	13,36
19,05	3,0	0,5303	0,0278	13,87
19,13	3,4	0,5587	0,0292	14,30
19,20	3,8	0,5877	0,0306	14,72
19,42	4,2	0,6188	0,0319	15,28
19,03	4,6	0,6436	0,0338	15,27
18,61	5,0	0,6682	0,0359	15,21
18,37	5,4	0,6953	0,0379	15,32
18,28	5,8	0,7245	0,0396	15,56
17,33	6,2	0,7414	0,0428	14,92
16,47	6,6	0,7581	0,0460	14,34
16,11	7,0	0,7821	0,0486	14,24
16,24	7,4	0,8142	0,0502	14,65
12,25	8,2	0,7831	0,0639	10,84
11,41	9,0	0,8099	0,0710	10,27
9,15	9,8	0,7787	0,0851	8,07
9,34	10,2	0,8060	0,0863	8,39
16,35	11,8	1,0513	0,0643	16,76
14,38	12,2	1,0264	0,0714	14,57
12,47	12,6	0,9964	0,0799	12,45

12.1.3 **Airfoil ID: *S5010***

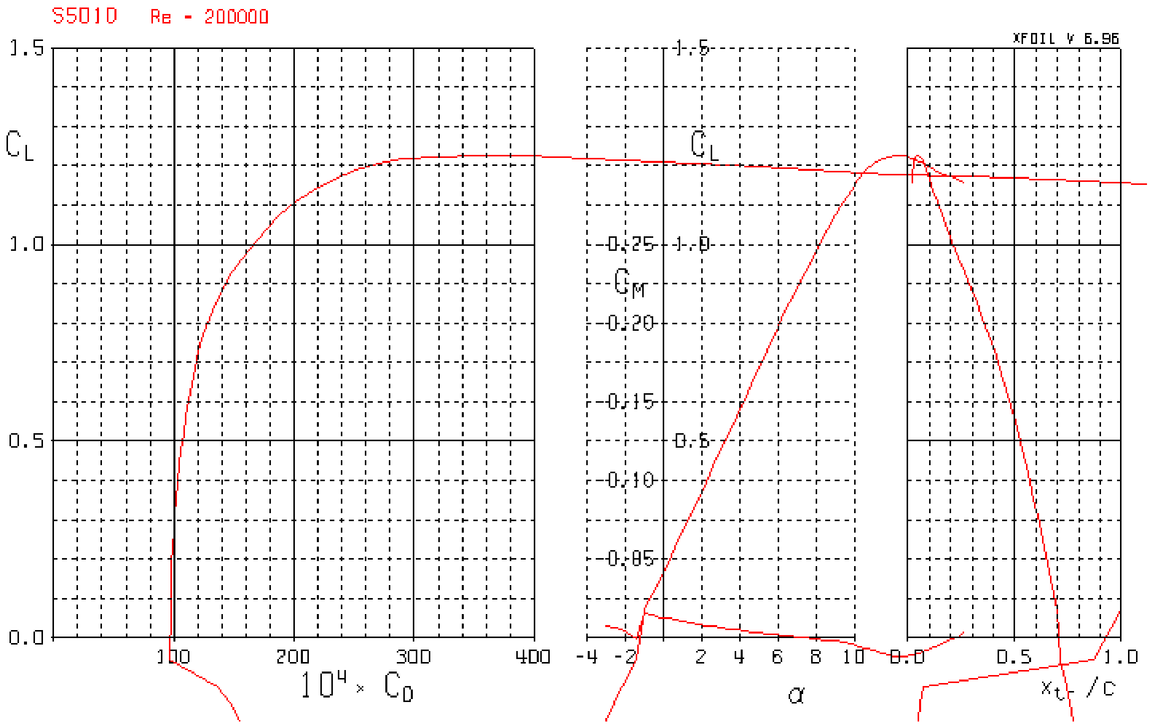




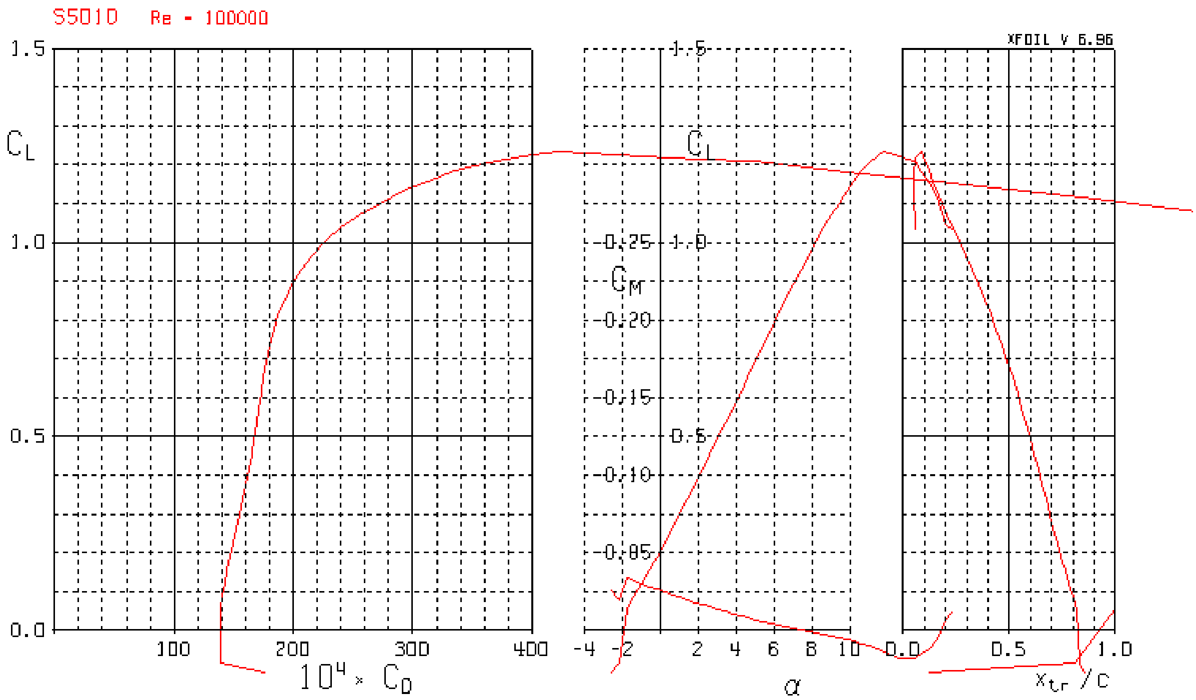
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
4,12	-0,6	0,0283	0,0069	0,69
18,20	-0,2	0,1287	0,0071	6,53
25,48	0,2	0,1799	0,0071	10,81
31,21	0,6	0,2213	0,0071	14,68
36,72	1,0	0,2629	0,0072	18,83
42,17	1,4	0,3049	0,0072	23,29
47,34	1,8	0,3470	0,0073	27,89
52,24	2,2	0,3892	0,0075	32,59
57,01	2,6	0,4316	0,0076	37,46
61,32	3,0	0,4740	0,0077	42,22
65,39	3,4	0,5166	0,0079	47,00
69,11	3,8	0,5591	0,0081	51,68
72,28	4,2	0,6014	0,0083	56,06
75,03	4,6	0,6438	0,0086	60,21
77,25	5,0	0,6860	0,0089	63,98
78,95	5,4	0,7279	0,0092	67,36
79,92	5,8	0,7696	0,0096	70,11
80,63	6,2	0,8111	0,0101	72,61
80,54	6,6	0,8521	0,0106	74,34
80,15	7,0	0,8929	0,0111	75,74
79,42	7,4	0,9332	0,0118	76,72
78,28	7,8	0,9730	0,0124	77,21
76,67	8,2	1,0120	0,0132	77,13
74,85	8,6	1,0502	0,0140	76,71
72,95	9,0	1,0877	0,0149	76,08
70,88	9,4	1,1241	0,0159	75,15
68,55	9,8	1,1592	0,0169	73,81
65,95	10,2	1,1923	0,0181	72,01
63,45	10,6	1,2239	0,0193	70,19
60,43	11,0	1,2521	0,0207	67,62
57,27	11,4	1,2766	0,0223	64,71
53,97	11,8	1,2963	0,0240	61,44
48,01	12,2	1,2948	0,0270	54,63
41,51	12,6	1,2894	0,0311	47,14
36,46	13,0	1,2881	0,0353	41,38
31,34	13,4	1,2787	0,0408	35,44
27,21	13,8	1,2692	0,0467	30,65
24,07	14,2	1,2626	0,0525	27,05
20,78	14,6	1,2455	0,0600	23,19
18,28	15,0	1,2319	0,0674	20,29
15,87	15,4	1,2118	0,0764	17,47
14,02	15,8	1,1942	0,0852	15,32



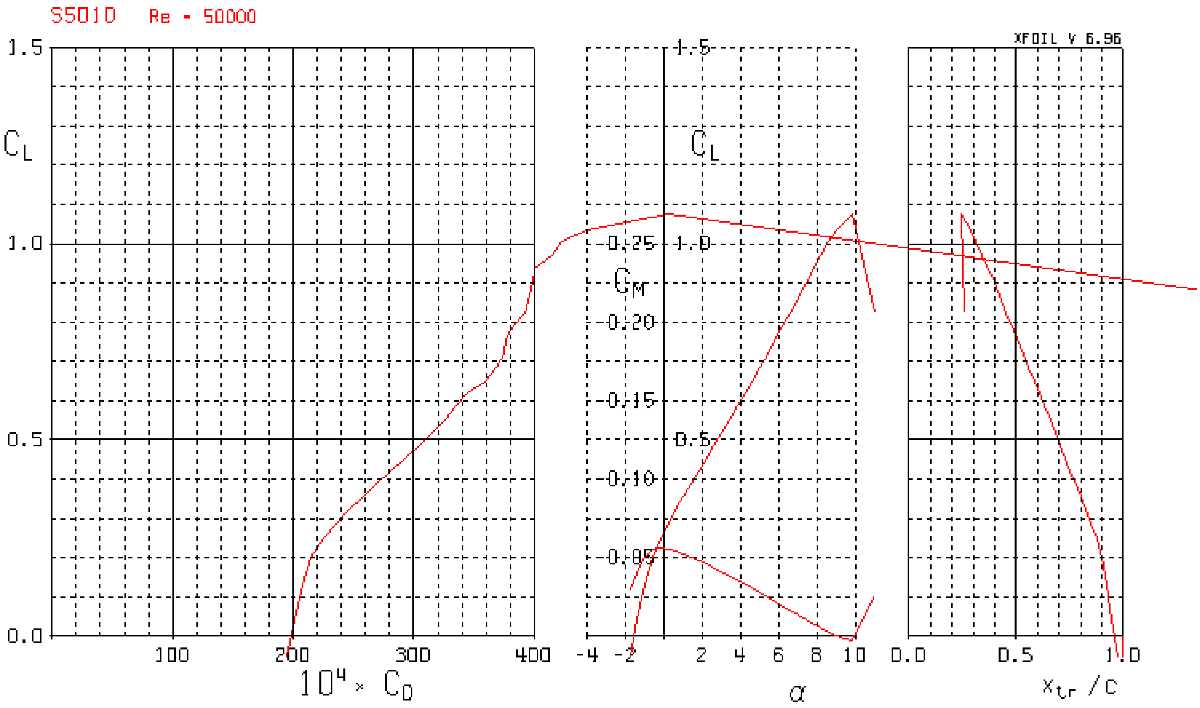
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
11,20	-0,6	0,0898	0,0080	3,36
17,68	-0,2	0,1414	0,0080	6,65
22,75	0,2	0,1822	0,0080	9,71
27,75	0,6	0,2234	0,0081	13,12
32,57	1,0	0,2648	0,0081	16,76
37,34	1,4	0,3066	0,0082	20,68
41,94	1,8	0,3485	0,0083	24,76
46,27	2,2	0,3905	0,0084	28,91
50,55	2,6	0,4327	0,0086	33,25
54,46	3,0	0,4749	0,0087	37,53
58,11	3,4	0,5172	0,0089	41,79
61,56	3,8	0,5596	0,0091	46,05
64,64	4,2	0,6018	0,0093	50,15
67,28	4,6	0,6439	0,0096	53,99
69,56	5,0	0,6859	0,0099	57,61
71,26	5,4	0,7276	0,0102	60,79
72,42	5,8	0,7691	0,0106	63,51
73,28	6,2	0,8105	0,0111	65,97
73,39	6,6	0,8513	0,0116	67,71
73,16	7,0	0,8918	0,0122	69,09
72,43	7,4	0,9315	0,0129	69,91
71,32	7,8	0,9707	0,0136	70,27
70,04	8,2	1,0093	0,0144	70,37
68,37	8,6	1,0468	0,0153	69,96
66,45	9,0	1,0831	0,0163	69,15
64,25	9,4	1,1179	0,0174	67,93
61,55	9,8	1,1504	0,0187	66,02
59,10	10,2	1,1814	0,0200	64,24
56,82	10,6	1,2109	0,0213	62,53
54,08	11,0	1,2362	0,0229	60,13
50,87	11,4	1,2559	0,0247	57,00
47,04	11,8	1,2648	0,0269	52,90
42,42	12,2	1,2689	0,0299	47,79
36,91	12,6	1,2643	0,0343	41,51
31,87	13,0	1,2566	0,0394	35,72
27,73	13,4	1,2480	0,0450	30,98
24,10	13,8	1,2363	0,0513	26,80
20,71	14,2	1,2173	0,0588	22,85
18,45	14,6	1,2082	0,0655	20,28
16,45	15,0	1,1965	0,0727	17,99
14,74	15,4	1,1858	0,0805	16,05
13,29	15,8	1,1740	0,0884	14,39



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
7,28	-1,0	0,0710	0,0098	1,94
11,25	-0,6	0,1096	0,0097	3,73
15,23	-0,2	0,1488	0,0098	5,88
19,26	0,2	0,1891	0,0098	8,37
23,23	0,6	0,2297	0,0099	11,13
27,08	1,0	0,2703	0,0100	14,08
30,92	1,4	0,3117	0,0101	17,26
34,64	1,8	0,3530	0,0102	20,58
38,20	2,2	0,3946	0,0103	24,00
41,67	2,6	0,4363	0,0105	27,53
44,97	3,0	0,4780	0,0106	31,09
48,09	3,4	0,5198	0,0108	34,67
51,05	3,8	0,5616	0,0110	38,26
53,78	4,2	0,6034	0,0112	41,77
56,25	4,6	0,6452	0,0115	45,18
58,45	5,0	0,6868	0,0118	48,44
60,23	5,4	0,7282	0,0121	51,40
61,59	5,8	0,7693	0,0125	54,02
62,62	6,2	0,8103	0,0129	56,37
63,00	6,6	0,8505	0,0135	58,10
63,07	7,0	0,8905	0,0141	59,51
62,50	7,4	0,9294	0,0149	60,25
61,72	7,8	0,9678	0,0157	60,72
58,87	8,6	1,0408	0,0177	60,06
57,11	9,0	1,0753	0,0188	59,22
55,00	9,4	1,1077	0,0201	57,89
52,57	9,8	1,1376	0,0216	56,07
50,00	10,2	1,1645	0,0233	53,96
47,32	10,6	1,1881	0,0251	51,57
44,77	11,0	1,2075	0,0270	49,20
41,89	11,4	1,2173	0,0291	46,22
34,90	12,2	1,2271	0,0352	38,66
31,18	12,6	1,2262	0,0393	34,52
27,44	13,0	1,2199	0,0445	30,31
24,06	13,4	1,2092	0,0503	26,46
21,33	13,8	1,2011	0,0563	23,38
16,89	14,6	1,1784	0,0698	18,34
15,37	15,0	1,1720	0,0762	16,64
14,05	15,4	1,1641	0,0829	15,16
12,65	15,8	1,1521	0,0911	13,57

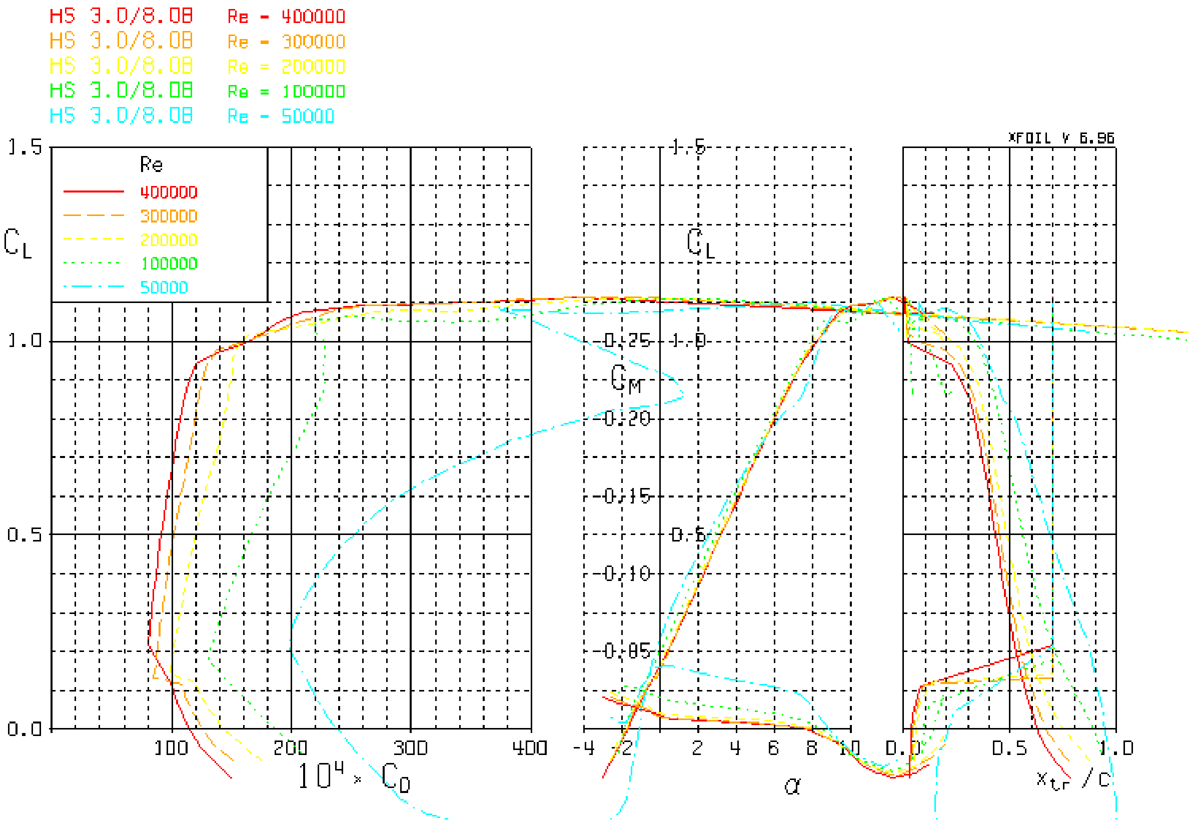


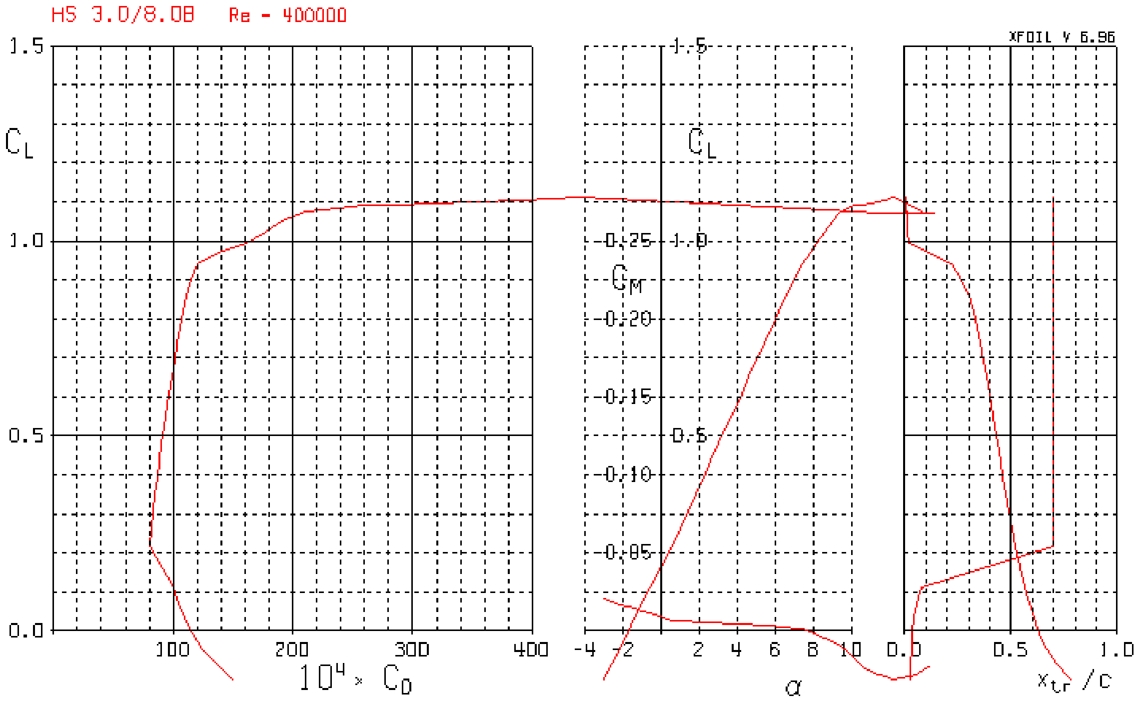
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
3,92	-1,8	0,0546	0,0139	0,92
6,17	-1,4	0,0867	0,0141	1,82
8,41	-1,0	0,1199	0,0143	2,91
10,64	-0,6	0,1540	0,0145	4,18
12,92	-0,2	0,1903	0,0147	5,64
15,08	0,2	0,2258	0,0150	7,17
17,27	0,6	0,2636	0,0153	8,87
19,38	1,0	0,3002	0,0155	10,62
21,46	1,4	0,3389	0,0158	12,49
23,55	1,8	0,3764	0,0160	14,45
25,55	2,2	0,4157	0,0163	16,47
27,56	2,6	0,4544	0,0165	18,58
29,59	3,0	0,4936	0,0167	20,79
31,52	3,4	0,5333	0,0169	23,02
33,47	3,8	0,5730	0,0171	25,34
35,45	4,2	0,6129	0,0173	27,75
37,36	4,6	0,6531	0,0175	30,19
39,08	5,0	0,6933	0,0177	32,54
40,68	5,4	0,7335	0,0180	34,84
42,18	5,8	0,7736	0,0183	37,10
43,48	6,2	0,8136	0,0187	39,22
44,28	6,6	0,8528	0,0193	40,89
44,85	7,0	0,8917	0,0199	42,36
45,07	7,4	0,9298	0,0206	43,46
44,78	7,8	0,9667	0,0216	44,02
44,17	8,2	1,0026	0,0227	44,22
43,22	8,6	1,0372	0,0240	44,01
41,97	9,0	1,0706	0,0255	43,42
40,44	9,4	1,1016	0,0272	42,45
38,81	9,8	1,1310	0,0291	41,28
37,05	10,2	1,1589	0,0313	39,88
35,32	10,6	1,1850	0,0336	38,45
33,14	11,0	1,2058	0,0364	36,40
31,14	11,4	1,2224	0,0393	34,42
29,28	11,8	1,2350	0,0422	32,54
24,77	12,6	1,2223	0,0493	27,39
22,49	13,0	1,2125	0,0539	24,77
20,63	13,4	1,2095	0,0586	22,69
17,77	13,8	1,1797	0,0664	19,30
15,32	14,2	1,1511	0,0751	16,44
12,97	14,6	1,1150	0,0860	13,69
9,96	15,0	1,0467	0,1051	10,19
8,93	15,4	1,0290	0,1153	9,06



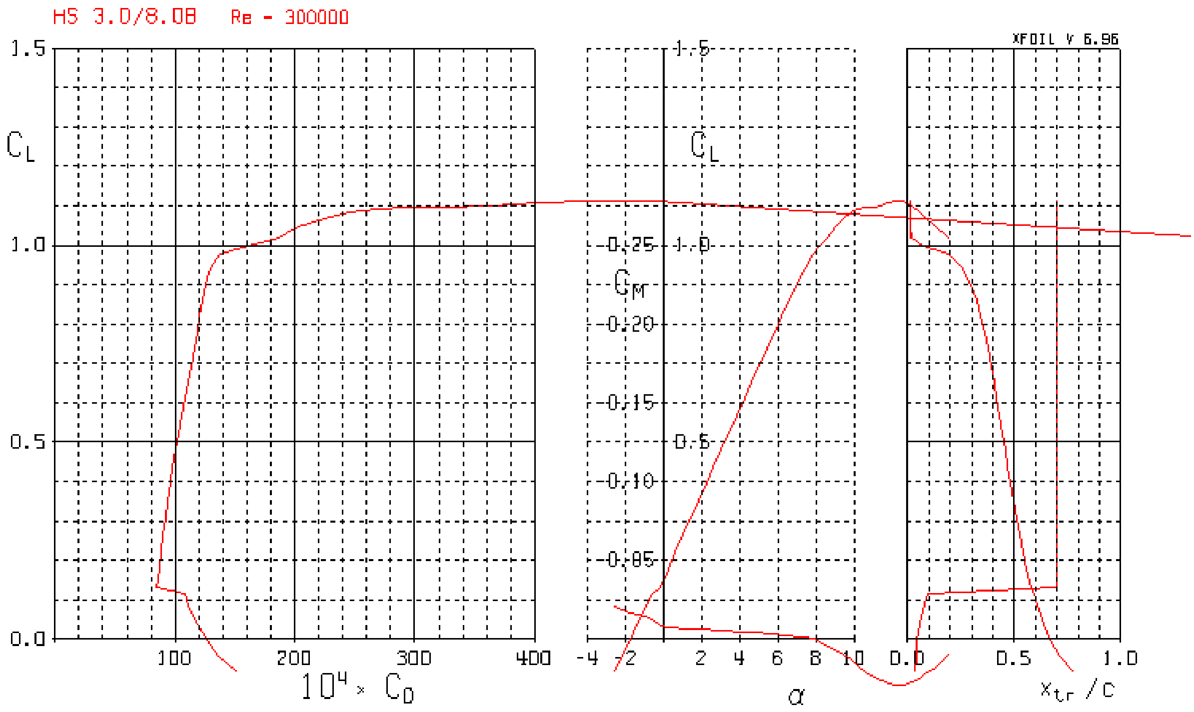
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
2,12	-1,4	0,0426	0,0201	0,44
6,37	-1,0	0,1324	0,0208	2,32
9,33	-0,6	0,2006	0,0215	4,18
11,03	-0,2	0,2478	0,0225	5,49
12,20	0,2	0,2869	0,0235	6,53
13,12	0,6	0,3229	0,0246	7,45
13,85	1,0	0,3569	0,0258	8,28
14,47	1,4	0,3896	0,0269	9,03
15,02	1,8	0,4219	0,0281	9,76
15,52	2,2	0,4540	0,0293	10,45
15,99	2,6	0,4863	0,0304	11,15
16,46	3,0	0,5189	0,0315	11,86
16,94	3,4	0,5520	0,0326	12,59
17,46	3,8	0,5856	0,0335	13,36
18,03	4,2	0,6200	0,0344	14,20
18,09	4,6	0,6496	0,0359	14,58
18,55	5,0	0,6832	0,0368	15,33
19,20	5,4	0,7188	0,0374	16,28
20,10	5,8	0,7564	0,0376	17,48
20,66	6,2	0,7913	0,0383	18,38
21,06	6,6	0,8250	0,0392	19,13
22,51	7,4	0,8990	0,0399	21,35
23,45	7,8	0,9385	0,0400	22,72
23,41	8,2	0,9704	0,0415	23,06
23,83	8,6	1,0072	0,0423	23,92
23,34	9,0	1,0362	0,0444	23,76
22,18	9,4	1,0568	0,0477	22,80
21,02	9,8	1,0756	0,0512	21,80
7,56	11,0	0,8234	0,1090	6,86

12.1.4 **Airfoil ID: *HS 3.0 / 8.0B***

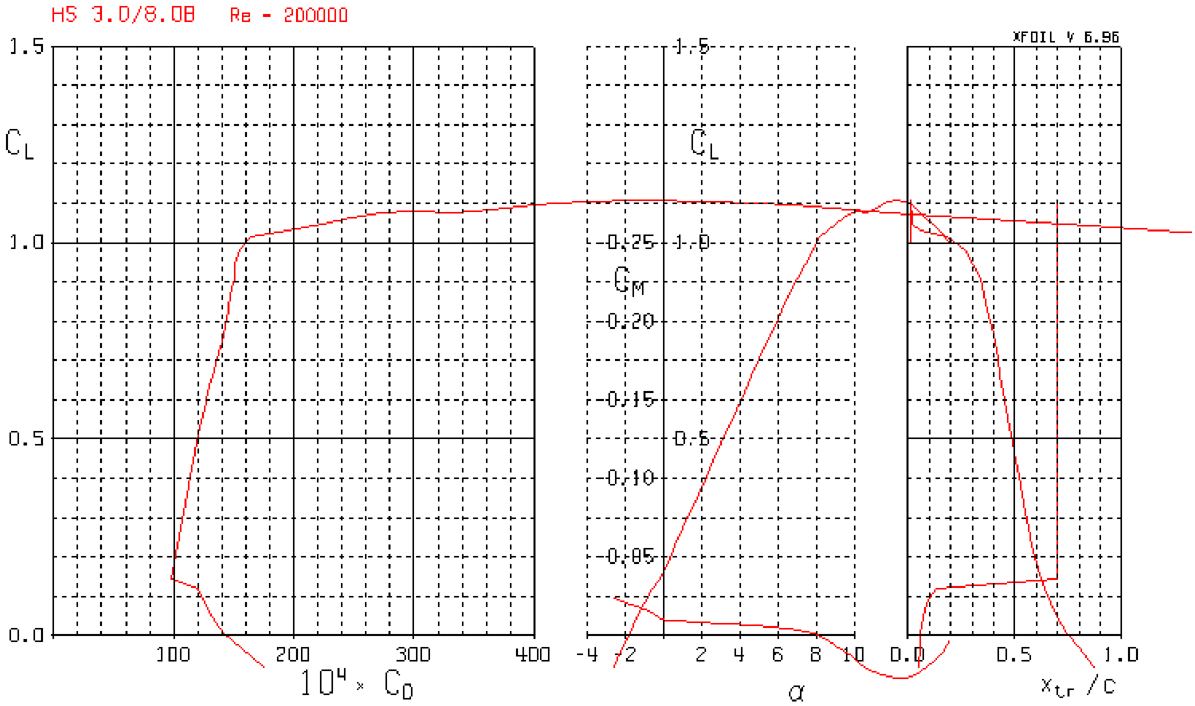




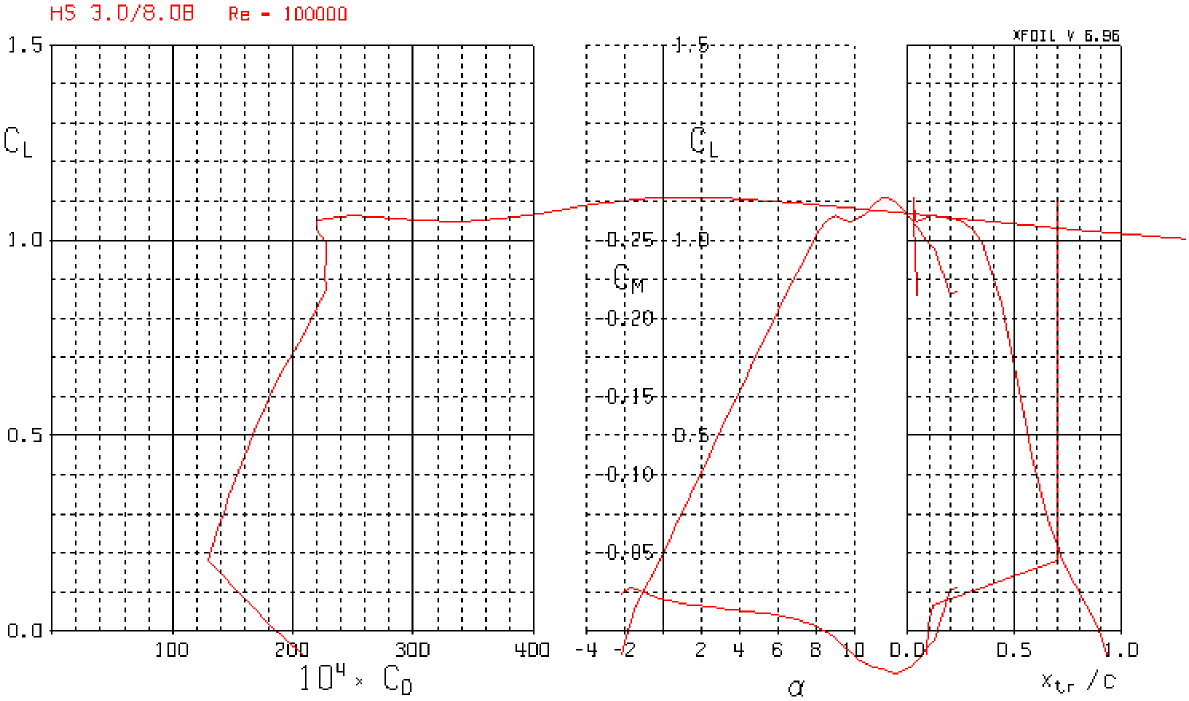
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
2,94	-1,4	0,0320	0,0109	0,53
6,88	-1,0	0,0716	0,0104	1,84
11,13	-0,6	0,1121	0,0101	3,73
27,22	0,6	0,2183	0,0080	12,72
32,07	1,0	0,2620	0,0082	16,41
36,69	1,4	0,3056	0,0083	20,28
41,14	1,8	0,3493	0,0085	24,32
45,31	2,2	0,3928	0,0087	28,39
49,23	2,6	0,4362	0,0089	32,52
53,05	3,0	0,4796	0,0090	36,74
56,52	3,4	0,5228	0,0093	40,87
62,85	4,2	0,6090	0,0097	49,05
65,62	4,6	0,6516	0,0099	52,97
68,54	5,0	0,6943	0,0101	57,11
71,10	5,4	0,7366	0,0104	61,02
73,52	5,8	0,7786	0,0106	64,87
75,69	6,2	0,8205	0,0108	68,56
77,64	6,6	0,8618	0,0111	72,08
78,42	7,0	0,9018	0,0115	74,47
77,94	7,4	0,9407	0,0121	75,59
70,21	7,8	0,9696	0,0138	69,13
61,49	8,2	0,9931	0,0162	61,28
57,74	8,6	1,0226	0,0177	58,39
55,13	9,0	1,0524	0,0191	56,55
51,11	9,4	1,0744	0,0210	52,98
45,47	9,8	1,0823	0,0238	47,31
41,95	10,2	1,0902	0,0260	43,80
38,20	10,6	1,0918	0,0286	39,92
34,57	11,0	1,0944	0,0317	36,16
31,16	11,4	1,0991	0,0353	32,67
28,15	11,8	1,1050	0,0393	29,59
25,26	12,2	1,1121	0,0440	26,64
17,47	13,4	1,0850	0,0621	18,20
15,51	13,8	1,0715	0,0691	16,06
14,55	14,2	1,0732	0,0738	15,07



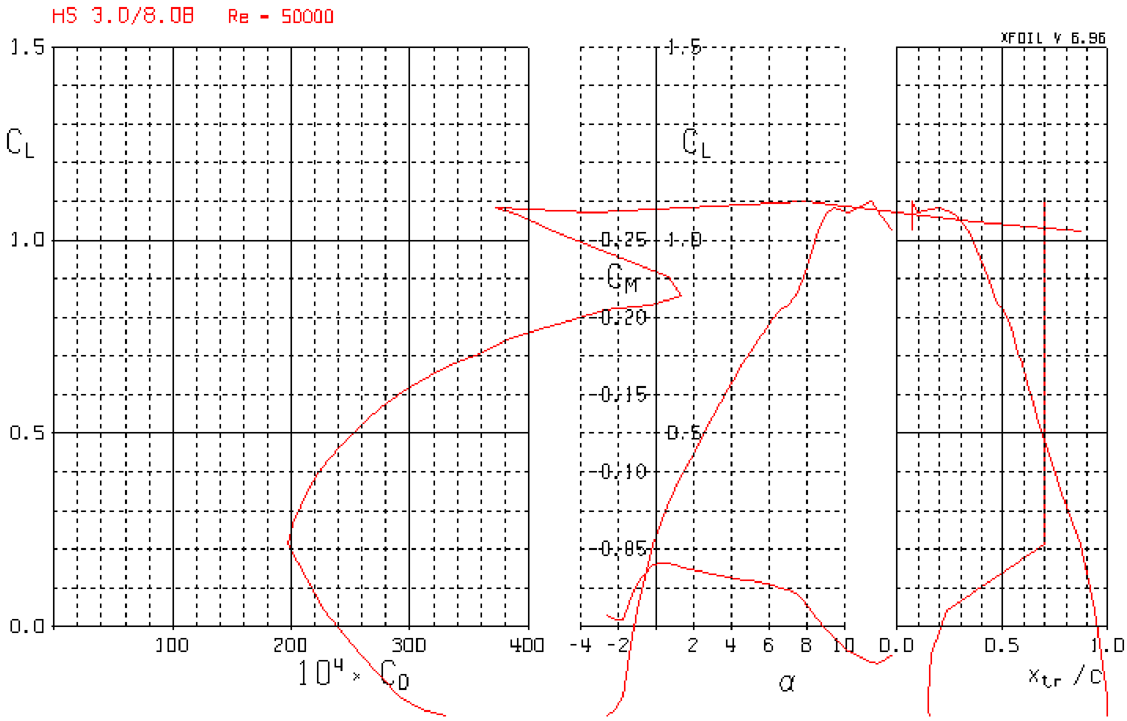
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
3,11	-1,4	0,0370	0,0119	0,60
6,72	-1,0	0,0755	0,0112	1,85
10,62	-0,6	0,1151	0,0108	3,60
15,66	-0,2	0,1333	0,0085	5,72
20,37	0,2	0,1768	0,0087	8,56
24,86	0,6	0,2203	0,0089	11,67
29,26	1,0	0,2639	0,0090	15,03
33,41	1,4	0,3074	0,0092	18,53
37,32	1,8	0,3508	0,0094	22,10
41,07	2,2	0,3943	0,0096	25,79
44,56	2,6	0,4376	0,0098	29,48
47,89	3,0	0,4808	0,0100	33,21
50,96	3,4	0,5239	0,0103	36,89
53,73	3,8	0,5669	0,0106	40,46
56,46	4,2	0,6098	0,0108	44,09
58,83	4,6	0,6524	0,0111	47,52
61,32	5,0	0,6947	0,0113	51,11
63,42	5,4	0,7369	0,0116	54,44
65,82	5,8	0,7787	0,0118	58,09
68,05	6,2	0,8200	0,0121	61,62
69,97	6,6	0,8613	0,0123	64,93
71,77	7,0	0,9014	0,0126	68,14
72,37	7,4	0,9401	0,0130	70,17
70,97	7,8	0,9765	0,0138	70,13
61,93	8,2	0,9995	0,0161	61,91
54,73	8,6	1,0208	0,0187	55,30
51,65	9,0	1,0484	0,0203	52,88
47,37	9,4	1,0672	0,0225	48,93
44,26	9,8	1,0852	0,0245	46,10
40,76	10,2	1,0932	0,0268	42,62
37,08	10,6	1,0942	0,0295	38,79
33,60	11,0	1,0966	0,0326	35,18
30,35	11,4	1,1009	0,0363	31,85
27,53	11,8	1,1076	0,0402	28,98
24,87	12,2	1,1124	0,0447	26,23
22,16	12,6	1,1109	0,0501	23,36
19,60	13,0	1,1014	0,0562	20,57
16,92	13,4	1,0848	0,0641	17,62
14,79	13,8	1,0662	0,0721	15,27
13,11	14,2	1,0526	0,0803	13,45
10,25	15,0	1,0147	0,0990	10,33



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,51	-1,8	0,0072	0,0142	0,04
3,45	-1,4	0,0457	0,0133	0,74
6,63	-1,0	0,0833	0,0126	1,91
10,08	-0,6	0,1207	0,0120	3,50
14,51	-0,2	0,1423	0,0098	5,47
18,45	0,2	0,1856	0,0101	7,95
22,22	0,6	0,2289	0,0103	10,63
25,81	1,0	0,2723	0,0106	13,47
29,24	1,4	0,3158	0,0108	16,43
32,48	1,8	0,3592	0,0111	19,46
35,57	2,2	0,4026	0,0113	22,57
38,47	2,6	0,4459	0,0116	25,69
41,15	3,0	0,4893	0,0119	28,79
43,68	3,4	0,5324	0,0122	31,87
45,93	3,8	0,5755	0,0125	34,84
48,00	4,2	0,6183	0,0129	37,75
49,89	4,6	0,6611	0,0133	40,57
51,65	5,0	0,7035	0,0136	43,32
53,37	5,4	0,7456	0,0140	46,09
54,97	5,8	0,7871	0,0143	48,76
56,90	6,2	0,8279	0,0146	51,77
58,89	6,6	0,8680	0,0147	54,86
60,43	7,0	0,9082	0,0150	57,59
62,80	7,4	0,9458	0,0151	61,08
63,75	7,8	0,9831	0,0154	63,21
62,17	8,2	1,0159	0,0163	62,66
53,06	8,6	1,0315	0,0194	53,89
46,83	9,0	1,0476	0,0224	47,93
42,94	9,4	1,0644	0,0248	44,30
39,50	9,8	1,0765	0,0273	40,99
36,13	10,2	1,0793	0,0299	37,54
32,22	10,6	1,0763	0,0334	33,43
29,76	11,0	1,0853	0,0365	31,00
27,38	11,4	1,0949	0,0400	28,65
25,06	11,8	1,1055	0,0441	26,35
22,68	12,2	1,1103	0,0490	23,90
20,20	12,6	1,1062	0,0548	21,24
17,74	13,0	1,0944	0,0617	18,56
15,49	13,4	1,0768	0,0695	16,07
13,61	13,8	1,0585	0,0778	14,00
12,10	14,2	1,0421	0,0861	12,35
10,67	14,6	1,0212	0,0957	10,78
9,27	15,0	0,9934	0,1071	9,24

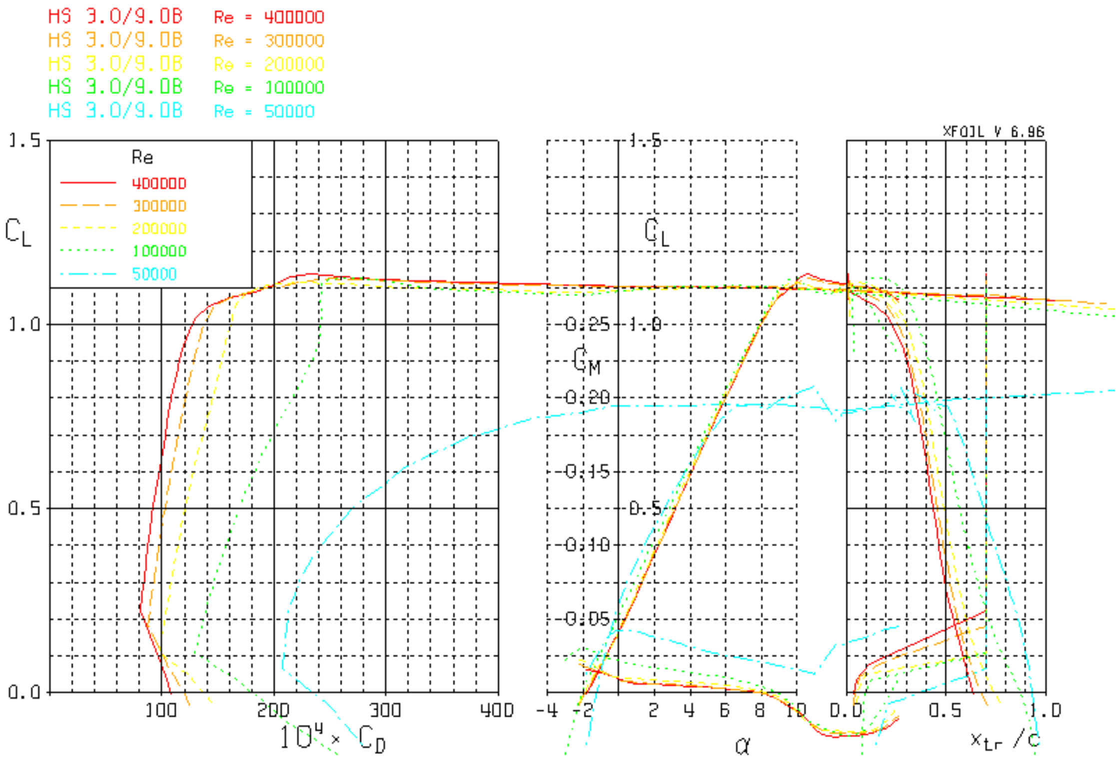


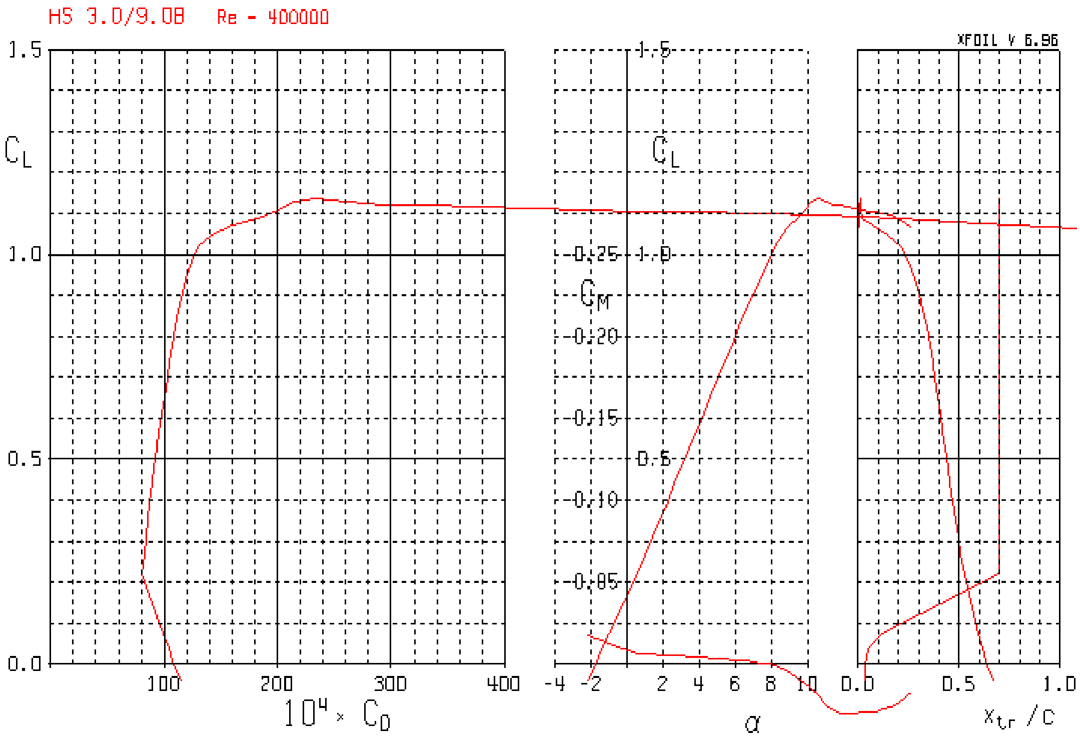
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,74	-1,8	0,0134	0,0182	0,09
3,91	-1,4	0,0645	0,0165	0,99
13,84	-0,2	0,1799	0,0130	5,87
16,52	0,2	0,2212	0,0134	7,77
19,02	0,6	0,2626	0,0138	9,74
21,33	1,0	0,3042	0,0143	11,77
23,48	1,4	0,3458	0,0147	13,80
25,46	1,8	0,3875	0,0152	15,85
27,27	2,2	0,4292	0,0157	17,86
28,91	2,6	0,4707	0,0163	19,84
30,42	3,0	0,5123	0,0168	21,77
31,80	3,4	0,5539	0,0174	23,66
32,99	3,8	0,5951	0,0180	25,45
33,98	4,2	0,6361	0,0187	27,10
34,98	4,6	0,6776	0,0194	28,80
35,60	5,0	0,7177	0,0202	30,16
36,38	5,4	0,7585	0,0209	31,68
37,11	5,8	0,7986	0,0215	33,16
37,64	6,2	0,8379	0,0223	34,46
38,44	6,6	0,8773	0,0228	36,01
40,59	7,0	0,9177	0,0226	38,88
41,97	7,4	0,9549	0,0228	41,02
43,47	7,8	0,9925	0,0228	43,31
46,60	8,2	1,0257	0,0220	47,20
47,88	8,6	1,0528	0,0220	49,12
42,78	9,0	1,0626	0,0248	44,10
34,63	9,4	1,0496	0,0303	35,48
30,98	9,8	1,0493	0,0339	31,73
28,34	10,2	1,0537	0,0372	29,09
26,39	10,6	1,0683	0,0405	27,28
24,39	11,0	1,0935	0,0448	25,51
22,22	11,4	1,1087	0,0499	23,40
19,94	11,8	1,1072	0,0555	20,98
17,77	12,2	1,0950	0,0616	18,60
15,68	12,6	1,0762	0,0686	16,27
13,71	13,0	1,0528	0,0768	14,06
11,88	13,4	1,0247	0,0863	12,03
10,39	13,8	0,9993	0,0962	10,38
9,19	14,2	0,9783	0,1064	9,09
5,53	15,0	0,8594	0,1555	5,12
5,46	15,4	0,8705	0,1594	5,10



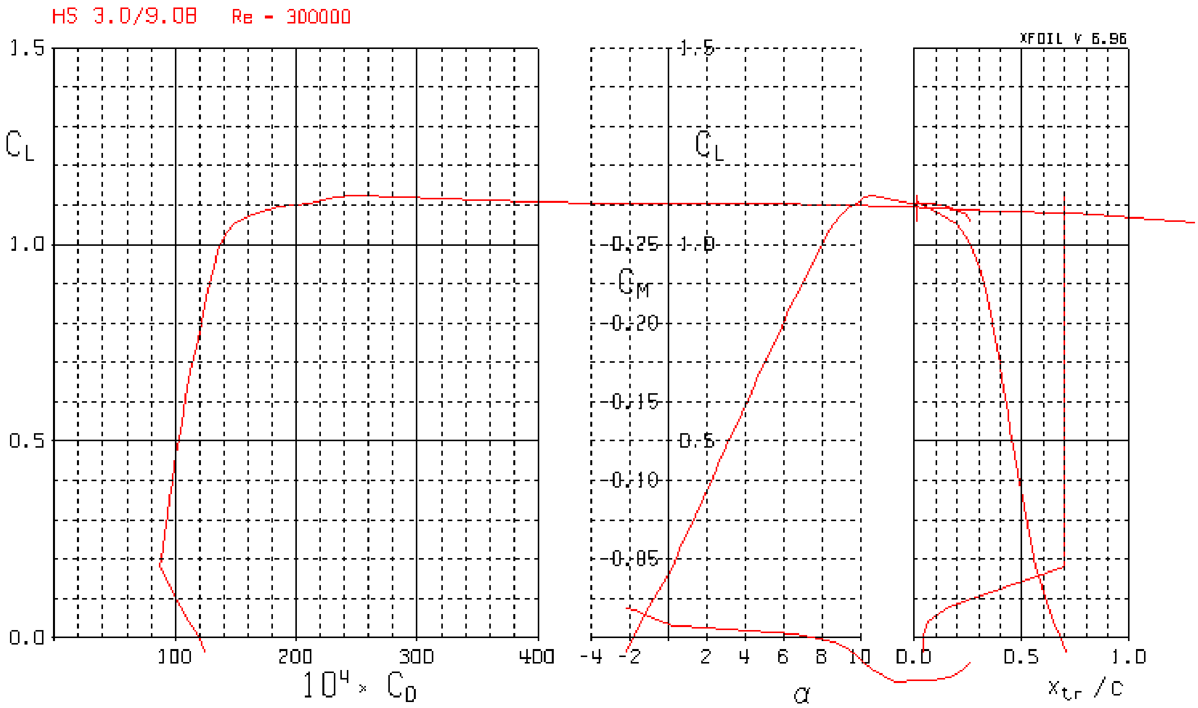
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
1,86	-1,0	0,0425	0,0228	0,38
10,77	-0,2	0,2125	0,0197	4,96
13,30	0,2	0,2677	0,0201	6,88
15,08	0,6	0,3125	0,0207	8,43
16,54	1,0	0,3542	0,0214	9,84
17,75	1,4	0,3942	0,0222	11,14
18,73	1,8	0,4329	0,0231	12,32
19,45	2,2	0,4702	0,0242	13,33
19,99	2,6	0,5072	0,0254	14,24
20,41	3,0	0,5440	0,0267	15,06
20,64	3,4	0,5800	0,0281	15,72
20,66	3,8	0,6148	0,0298	16,20
20,47	4,2	0,6481	0,0317	16,48
20,08	4,6	0,6796	0,0338	16,56
19,67	5,0	0,7105	0,0361	16,58
19,45	5,4	0,7433	0,0382	16,77
18,72	5,8	0,7688	0,0411	16,42
18,06	6,2	0,7934	0,0439	16,09
17,67	6,6	0,8205	0,0464	16,00
16,54	7,0	0,8335	0,0504	15,10
16,23	7,4	0,8580	0,0529	15,03
17,44	7,8	0,9046	0,0519	16,59
24,34	8,6	1,0248	0,0421	24,64
27,36	9,0	1,0664	0,0390	28,26
29,14	9,4	1,0835	0,0372	30,33
23,53	10,2	1,0697	0,0455	24,33
17,34	11,4	1,0985	0,0634	18,17
15,38	11,8	1,0758	0,0700	15,95
13,50	12,2	1,0488	0,0777	13,83
11,82	12,6	1,0249	0,0867	11,97

12.1.5 **Airfoil ID: *HS 3.0 / 9.0B***

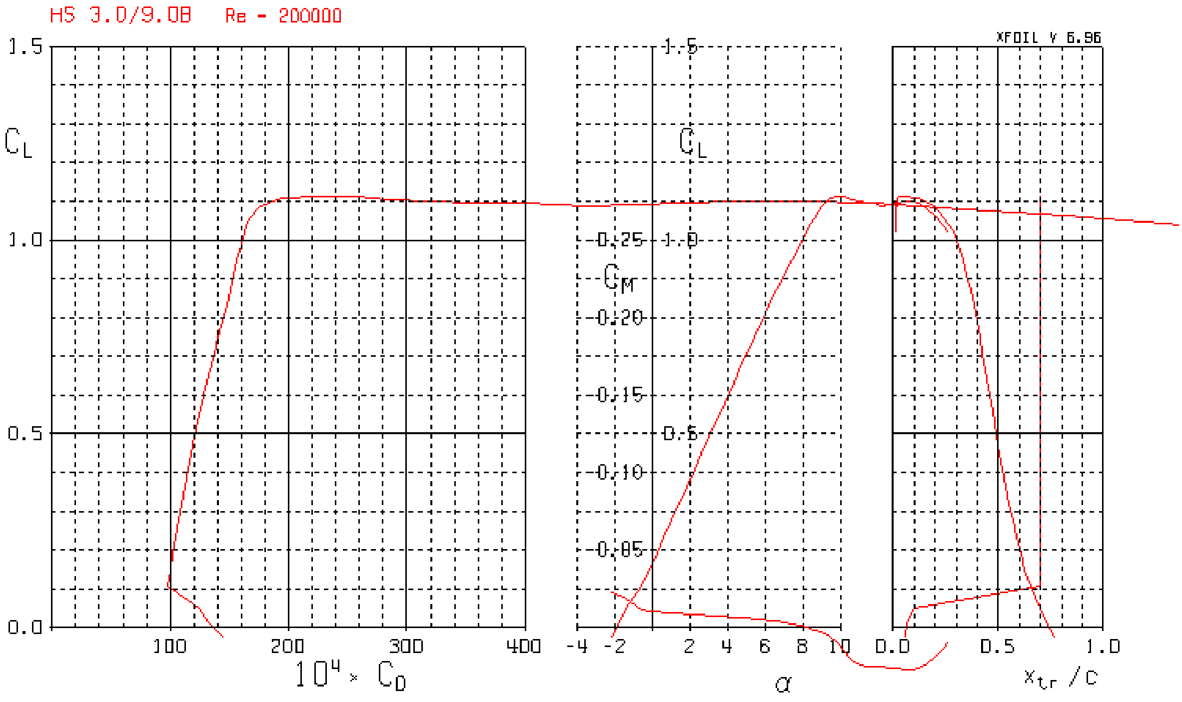




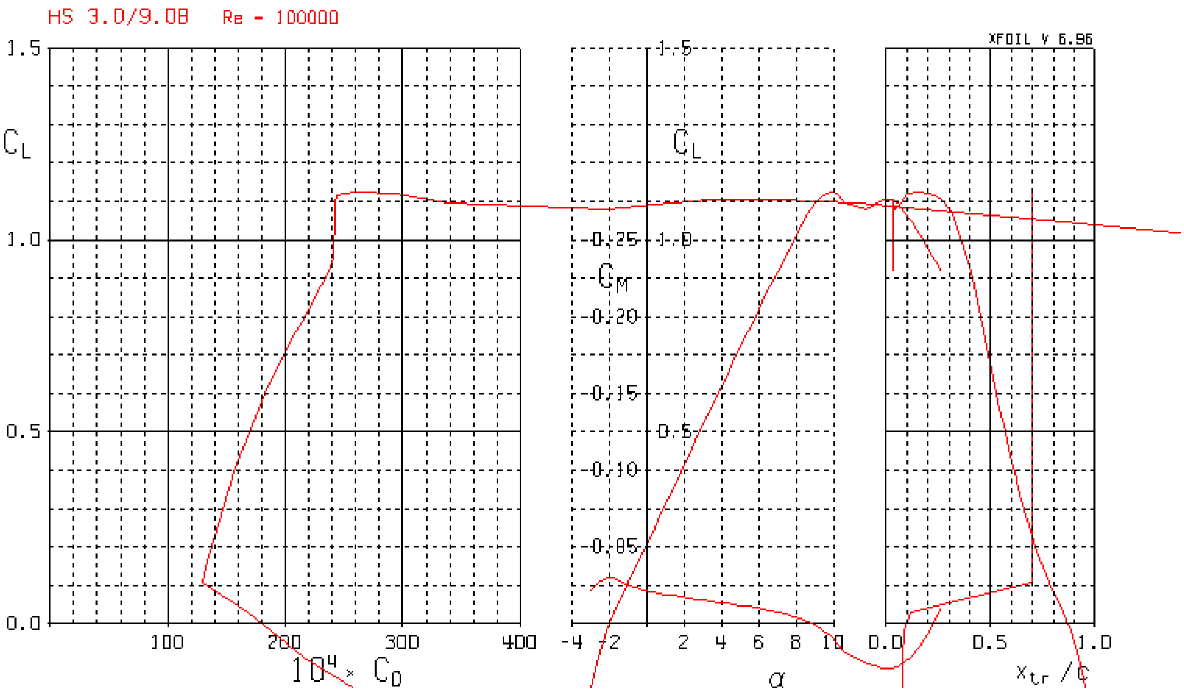
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
3,50	-1,4	0,0363	0,0104	0,67
7,59	-1,0	0,0751	0,0099	2,08
27,38	0,6	0,2218	0,0081	12,90
32,17	1,0	0,2654	0,0083	16,57
36,67	1,4	0,3088	0,0084	20,38
41,01	1,8	0,3523	0,0086	24,34
45,18	2,2	0,3958	0,0088	28,43
49,12	2,6	0,4391	0,0089	32,55
52,84	3,0	0,4824	0,0091	36,70
56,31	3,4	0,5254	0,0093	40,82
59,52	3,8	0,5684	0,0096	44,87
62,56	4,2	0,6112	0,0098	48,91
65,31	4,6	0,6538	0,0100	52,81
67,85	5,0	0,6961	0,0103	56,61
70,46	5,4	0,7384	0,0105	60,54
72,86	5,8	0,7803	0,0107	64,36
74,77	6,2	0,8217	0,0110	67,78
76,69	6,6	0,8628	0,0113	71,24
77,99	7,0	0,9031	0,0116	74,11
79,18	7,4	0,9430	0,0119	76,89
79,35	7,8	0,9815	0,0124	78,61
78,34	8,2	1,0184	0,0130	79,06
73,99	8,6	1,0492	0,0142	75,79
66,80	9,0	1,0722	0,0161	69,17
59,50	9,4	1,0889	0,0183	62,09
55,28	9,8	1,1089	0,0201	58,21
52,58	10,2	1,1300	0,0215	55,90
48,81	10,6	1,1377	0,0233	52,06
43,34	11,0	1,1287	0,0260	46,05
38,30	11,4	1,1219	0,0293	40,57
33,85	11,8	1,1188	0,0331	35,81
29,71	12,2	1,1148	0,0375	31,37
26,28	12,6	1,1107	0,0423	27,69
23,57	13,0	1,1082	0,0470	24,82
21,27	13,4	1,1048	0,0520	22,35
19,34	13,8	1,1023	0,0570	20,31
17,71	14,2	1,1005	0,0622	18,58
16,09	14,6	1,0950	0,0681	16,84
14,56	15,0	1,0868	0,0747	15,18
13,14	15,4	1,0774	0,0820	13,64
11,76	15,8	1,0635	0,0904	12,13



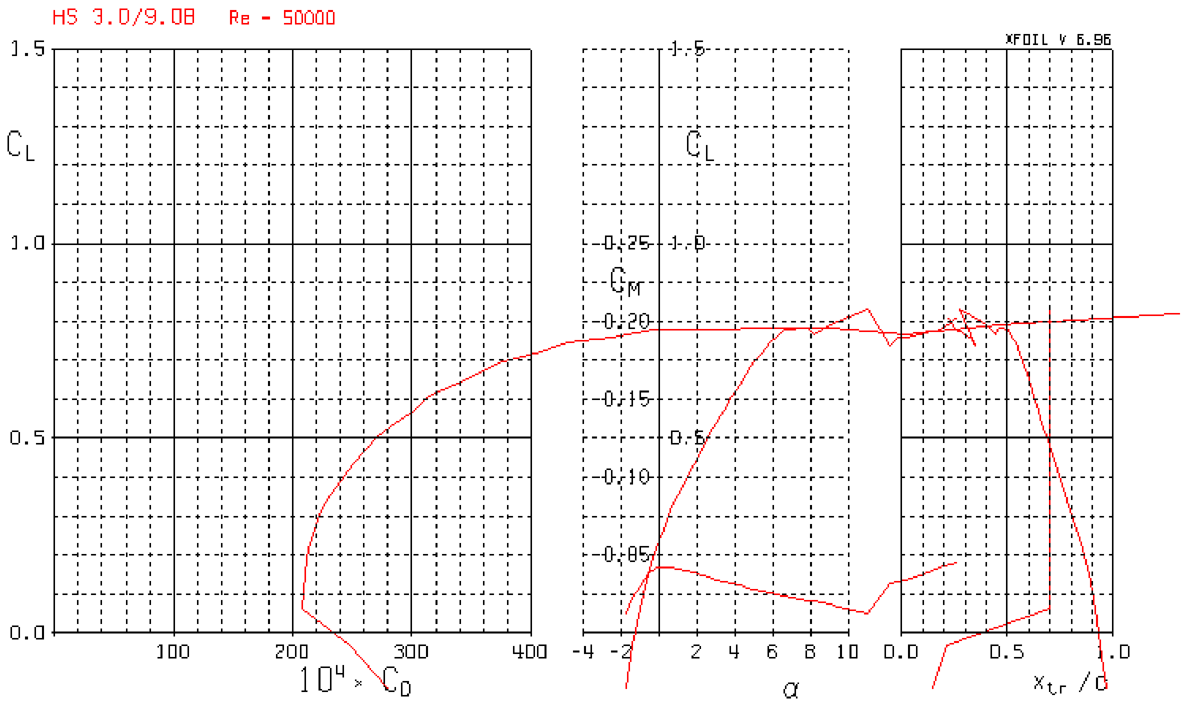
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,30	-1,8	0,0036	0,0119	0,02
3,66	-1,4	0,0408	0,0112	0,74
7,43	-1,0	0,0776	0,0105	2,07
20,68	0,2	0,1812	0,0088	8,81
25,07	0,6	0,2244	0,0090	11,88
29,36	1,0	0,2678	0,0091	15,20
33,38	1,4	0,3111	0,0093	18,62
37,27	1,8	0,3544	0,0095	22,19
40,96	2,2	0,3977	0,0097	25,83
44,44	2,6	0,4408	0,0099	29,50
47,67	3,0	0,4838	0,0102	33,15
50,64	3,4	0,5267	0,0104	36,75
53,52	3,8	0,5695	0,0106	40,39
56,22	4,2	0,6122	0,0109	43,99
58,72	4,6	0,6547	0,0112	47,51
61,03	5,0	0,6970	0,0114	50,95
63,16	5,4	0,7390	0,0117	54,30
65,10	5,8	0,7806	0,0120	57,52
66,91	6,2	0,8217	0,0123	60,66
68,82	6,6	0,8623	0,0125	63,91
70,38	7,0	0,9023	0,0128	66,86
71,45	7,4	0,9417	0,0132	69,34
72,61	7,8	0,9802	0,0135	71,89
72,72	8,2	1,0173	0,0140	73,34
71,26	8,6	1,0518	0,0148	73,08
65,84	9,0	1,0771	0,0164	68,33
59,04	9,4	1,0952	0,0186	61,79
52,27	9,8	1,1056	0,0212	54,97
48,44	10,2	1,1195	0,0231	51,25
44,90	10,6	1,1248	0,0251	47,62
31,51	11,8	1,1129	0,0353	33,24
27,73	12,2	1,1090	0,0400	29,20
24,70	12,6	1,1060	0,0448	25,98
22,30	13,0	1,1042	0,0495	23,44
20,31	13,4	1,1029	0,0543	21,33
18,63	13,8	1,1023	0,0592	19,56
17,16	14,2	1,1020	0,0642	18,01
15,59	14,6	1,0964	0,0703	16,33
13,92	15,0	1,0839	0,0779	14,49
12,82	15,4	1,0786	0,0842	13,31
11,20	15,8	1,0568	0,0944	11,51



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,97	-1,8	0,0129	0,0133	0,11
3,94	-1,4	0,0488	0,0124	0,87
10,75	-0,6	0,1051	0,0098	3,48
14,80	-0,2	0,1480	0,0100	5,69
18,69	0,2	0,1910	0,0102	8,17
22,42	0,6	0,2341	0,0104	10,85
25,94	1,0	0,2773	0,0107	13,66
29,32	1,4	0,3205	0,0109	16,60
32,47	1,8	0,3637	0,0112	19,58
35,53	2,2	0,4068	0,0115	22,66
38,35	2,6	0,4499	0,0117	25,73
40,99	3,0	0,4931	0,0120	28,78
43,50	3,4	0,5359	0,0123	31,84
45,78	3,8	0,5787	0,0126	34,83
47,91	4,2	0,6214	0,0130	37,77
49,85	4,6	0,6640	0,0133	40,62
51,56	5,0	0,7064	0,0137	43,34
53,18	5,4	0,7482	0,0141	46,00
54,69	5,8	0,7897	0,0144	48,60
56,21	6,2	0,8308	0,0148	51,24
57,55	6,6	0,8713	0,0151	53,72
59,23	7,0	0,9104	0,0154	56,52
60,59	7,4	0,9489	0,0157	59,03
61,64	7,8	0,9868	0,0160	61,23
62,72	8,2	1,0229	0,0163	63,43
63,24	8,6	1,0568	0,0167	65,01
61,87	9,0	1,0876	0,0176	64,52
57,00	9,4	1,1080	0,0194	59,99
49,51	9,8	1,1140	0,0225	52,26
43,24	10,2	1,1116	0,0257	45,58
38,89	10,6	1,1061	0,0284	40,90
34,46	11,0	1,0990	0,0319	36,13
30,75	11,4	1,0970	0,0357	32,20
27,33	11,8	1,0944	0,0400	28,59
24,05	12,2	1,0867	0,0452	25,07
22,12	12,6	1,0902	0,0493	23,09
20,49	13,0	1,0953	0,0535	21,45
18,97	13,4	1,0993	0,0580	19,89
17,38	13,8	1,0996	0,0633	18,23
15,78	14,2	1,0946	0,0694	16,51
14,04	14,6	1,0810	0,0770	14,60
12,41	15,0	1,0626	0,0856	12,79
10,88	15,4	1,0400	0,0956	11,09
9,57	15,8	1,0173	0,1063	9,65

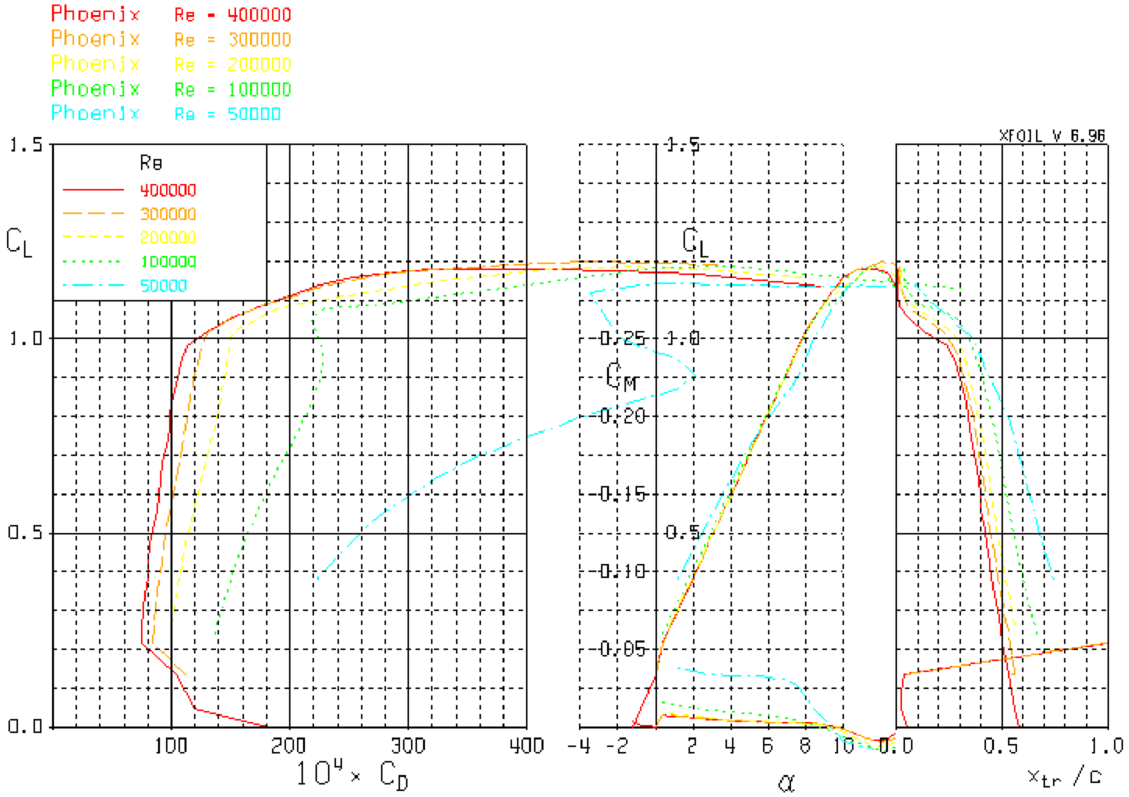


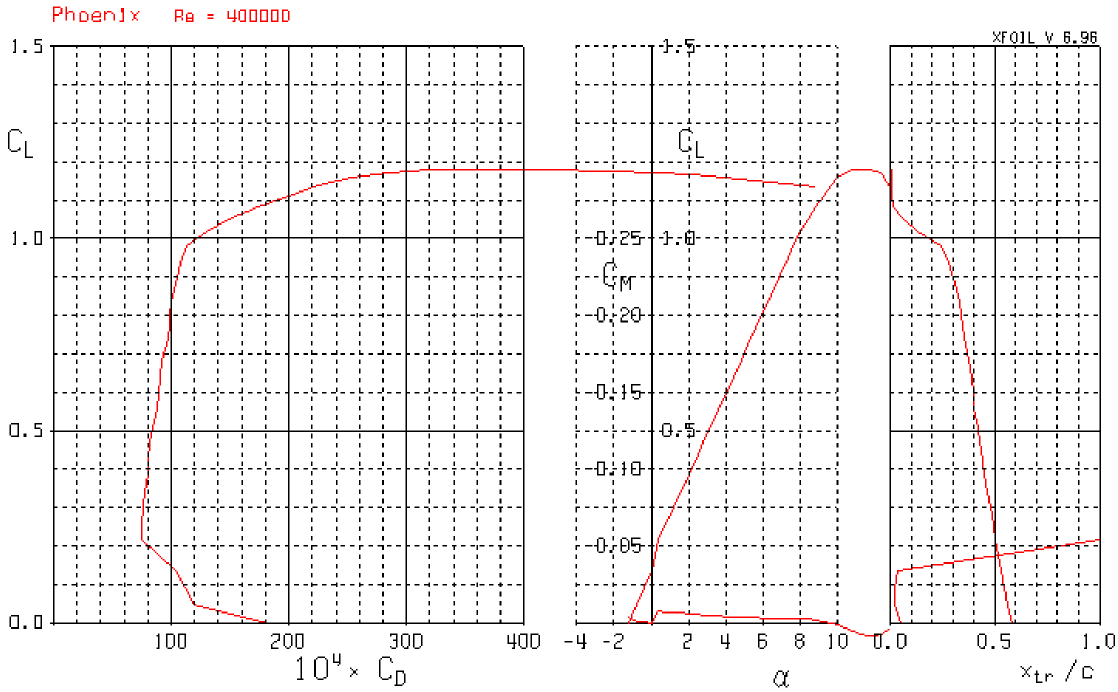
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
1,86	-1,8	0,0312	0,0168	0,33
8,25	-1,0	0,1071	0,0130	2,70
11,15	-0,6	0,1479	0,0133	4,29
13,87	-0,2	0,1886	0,0136	6,02
16,45	0,2	0,2296	0,0140	7,88
18,86	0,6	0,2706	0,0144	9,81
21,12	1,0	0,3117	0,0148	11,79
23,22	1,4	0,3529	0,0152	13,79
25,19	1,8	0,3942	0,0157	15,81
27,02	2,2	0,4355	0,0161	17,83
28,65	2,6	0,4765	0,0166	19,78
30,22	3,0	0,5179	0,0171	21,74
31,55	3,4	0,5587	0,0177	23,58
32,78	3,8	0,5996	0,0183	25,39
33,99	4,2	0,6411	0,0189	27,22
34,80	4,6	0,6807	0,0196	28,71
35,68	5,0	0,7211	0,0202	30,30
36,54	5,4	0,7622	0,0209	31,90
36,92	5,8	0,8005	0,0217	33,04
37,54	6,2	0,8401	0,0224	34,41
38,18	6,6	0,8796	0,0230	35,81
38,82	7,0	0,9178	0,0236	37,19
39,64	7,4	0,9556	0,0241	38,75
41,11	7,8	0,9933	0,0242	40,98
42,56	8,2	1,0300	0,0242	43,20
44,08	8,6	1,0659	0,0242	45,51
45,24	9,0	1,0961	0,0242	47,36
45,88	9,4	1,1166	0,0243	48,48
43,43	9,8	1,1266	0,0259	46,10
38,23	10,2	1,1220	0,0294	40,49
32,41	10,6	1,0969	0,0338	33,95
22,73	11,8	1,0791	0,0475	23,61
21,23	12,2	1,0921	0,0514	22,19
19,70	12,6	1,1042	0,0561	20,70
17,90	13,0	1,1057	0,0618	18,82
15,92	13,4	1,0943	0,0687	16,66
13,94	13,8	1,0729	0,0770	14,44
12,15	14,2	1,0470	0,0862	12,43
10,44	14,6	1,0146	0,0971	10,52
8,93	15,0	0,9800	0,1097	8,84
7,63	15,4	0,9456	0,1239	7,42
6,61	15,8	0,9161	0,1386	6,33



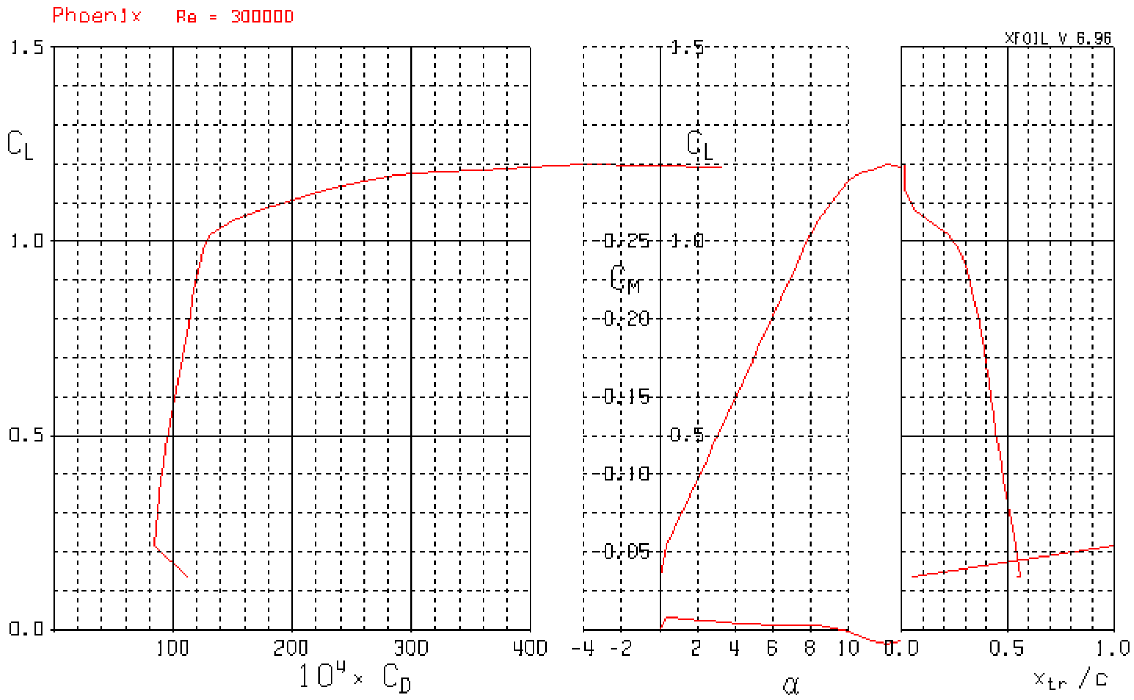
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
3,06	-1,0	0,0635	0,0208	0,77
7,13	-0,6	0,1501	0,0210	2,76
10,21	-0,2	0,2177	0,0213	4,76
12,23	0,2	0,2675	0,0219	6,33
14,06	0,6	0,3153	0,0224	7,90
15,30	1,0	0,3551	0,0232	9,12
16,32	1,4	0,3934	0,0241	10,24
17,19	1,8	0,4309	0,0251	11,29
17,94	2,2	0,4681	0,0261	12,28
18,63	2,6	0,5055	0,0271	13,24
18,86	3,0	0,5387	0,0286	13,84
18,95	3,4	0,5713	0,0301	14,33
19,37	3,8	0,6083	0,0314	15,11
19,04	4,2	0,6368	0,0335	15,19
18,78	4,6	0,6666	0,0355	15,34
18,52	5,0	0,6964	0,0376	15,46
17,78	5,4	0,7192	0,0405	15,08
17,31	5,8	0,7451	0,0430	14,94
16,20	6,2	0,7590	0,0468	14,12
15,48	6,6	0,7774	0,0502	13,65
14,14	7,0	0,7799	0,0552	12,49
13,00	7,4	0,7815	0,0601	11,49
12,00	7,8	0,7811	0,0651	10,61
10,72	8,2	0,7657	0,0714	9,38
10,34	8,6	0,7790	0,0754	9,12
10,00	9,0	0,7925	0,0793	8,90
9,58	9,4	0,8003	0,0836	8,57
8,25	11,0	0,8306	0,1006	7,52
5,31	12,2	0,7373	0,1389	4,56
5,20	12,6	0,7583	0,1459	4,53
4,99	13,0	0,7593	0,1520	4,35
4,68	13,8	0,7680	0,1642	4,10
4,55	14,2	0,7756	0,1705	4,01
4,42	14,6	0,7766	0,1756	3,90
4,32	15,0	0,7843	0,1815	3,83
4,24	15,4	0,7981	0,1884	3,79
4,16	15,8	0,8119	0,1951	3,75

12.1.6 **Airfoil ID: *Phoenix***

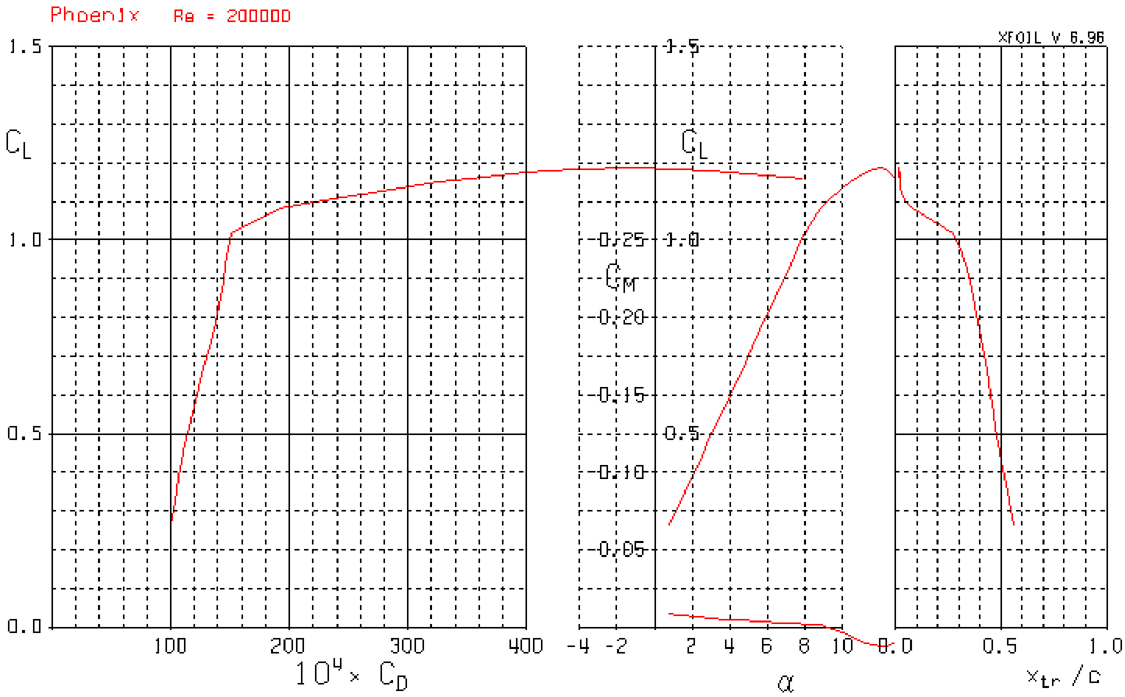




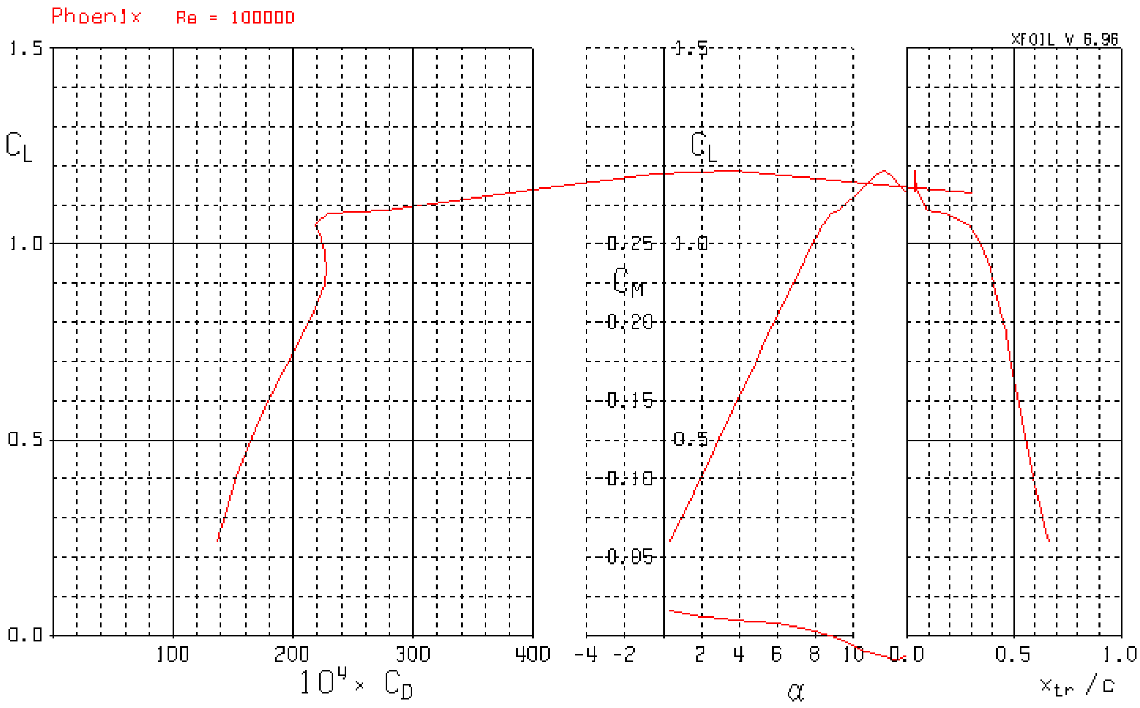
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,01	-1,2	0,0002	0,0182	0,00
3,93	-0,8	0,0471	0,0120	0,85
8,09	-0,4	0,0906	0,0112	2,43
12,98	0,0	0,1342	0,0103	4,75
28,61	0,4	0,2154	0,0075	13,28
34,19	0,8	0,2578	0,0075	17,36
39,43	1,2	0,3001	0,0076	21,60
44,00	1,6	0,3423	0,0078	25,74
48,01	2,0	0,3846	0,0080	29,78
52,83	2,4	0,4269	0,0081	34,52
57,15	2,8	0,4692	0,0082	39,15
60,81	3,2	0,5114	0,0084	43,49
62,53	3,6	0,5534	0,0089	46,52
66,74	4,0	0,5960	0,0089	51,52
70,59	4,4	0,6388	0,0091	56,42
73,77	4,8	0,6816	0,0092	60,90
75,89	5,2	0,7240	0,0095	64,57
77,77	5,6	0,7668	0,0099	68,10
81,68	6,0	0,8103	0,0099	73,53
84,16	6,4	0,8534	0,0101	77,75
84,99	6,8	0,8958	0,0105	80,44
86,73	7,2	0,9384	0,0108	84,01
86,50	7,6	0,9801	0,0113	85,64
78,23	8,0	1,0162	0,0130	78,86
70,84	8,4	1,0513	0,0148	72,64
62,21	8,8	1,0818	0,0174	64,70
55,42	9,2	1,1106	0,0200	58,40
50,55	9,6	1,1374	0,0225	53,91
45,71	10,0	1,1578	0,0253	49,18
41,12	10,4	1,1720	0,0285	44,52
36,84	10,8	1,1783	0,0320	39,99
32,58	11,2	1,1811	0,0363	35,41
28,44	11,6	1,1807	0,0415	30,90
24,69	12,0	1,1753	0,0476	26,77
21,41	12,4	1,1653	0,0544	23,12
17,54	12,8	1,1362	0,0648	18,70



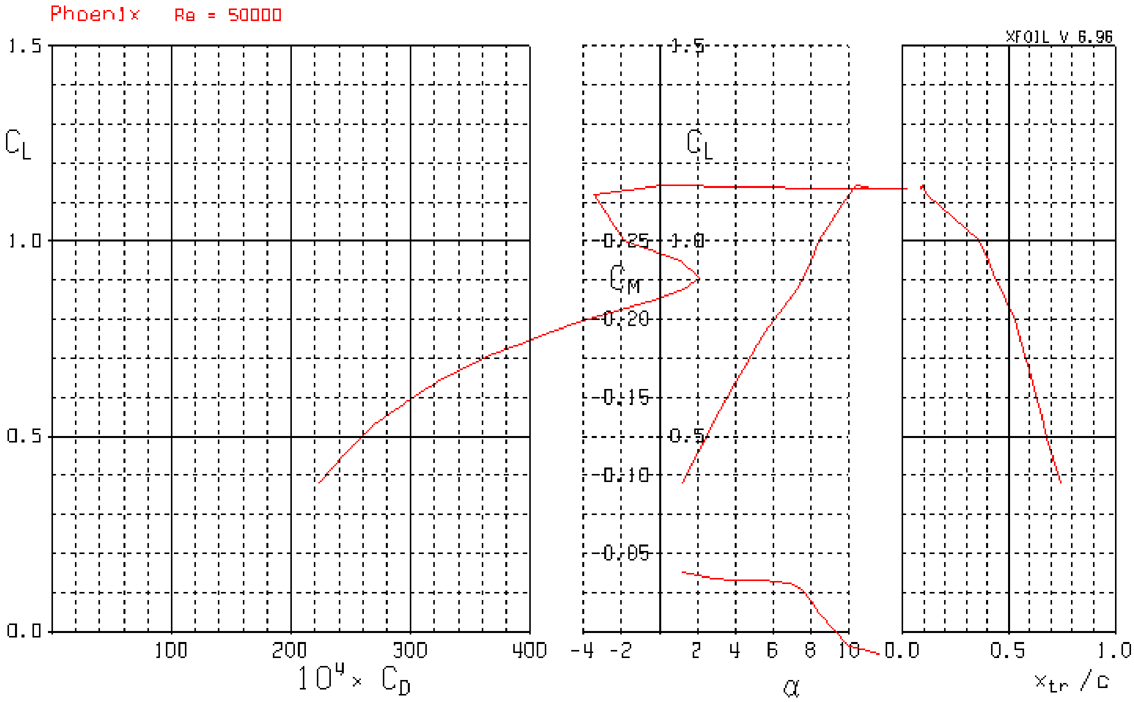
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
12,04	0,0	0,1351	0,0112	4,43
25,94	0,4	0,2171	0,0084	12,09
30,54	0,8	0,2593	0,0085	15,55
34,87	1,2	0,3013	0,0086	19,14
39,01	1,6	0,3433	0,0088	22,86
42,97	2,0	0,3854	0,0090	26,67
46,61	2,4	0,4274	0,0092	30,47
50,05	2,8	0,4695	0,0094	34,30
53,23	3,2	0,5115	0,0096	38,07
56,26	3,6	0,5536	0,0098	41,86
58,99	4,0	0,5958	0,0101	45,53
61,67	4,4	0,6383	0,0104	49,27
64,05	4,8	0,6808	0,0106	52,84
66,31	5,2	0,7234	0,0109	56,40
68,63	5,6	0,7659	0,0112	60,06
70,99	6,0	0,8086	0,0114	63,84
73,13	6,4	0,8512	0,0116	67,47
75,10	6,8	0,8937	0,0119	71,00
76,86	7,2	0,9361	0,0122	74,36
78,13	7,6	0,9782	0,0125	77,27
77,53	8,0	1,0188	0,0131	78,26
70,41	8,4	1,0533	0,0150	72,26
62,03	8,8	1,0837	0,0175	64,58
49,51	9,6	1,1362	0,0230	52,77
44,30	10,0	1,1541	0,0261	47,59
40,84	10,4	1,1724	0,0287	44,22
36,63	10,8	1,1786	0,0322	39,76
33,04	11,2	1,1850	0,0359	35,96
29,96	11,6	1,1932	0,0398	32,73
26,97	12,0	1,1980	0,0444	29,52
24,03	12,4	1,1969	0,0498	26,29
21,19	12,8	1,1894	0,0561	23,11



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
26,25	0,8	0,2635	0,0100	13,47
29,81	1,2	0,3053	0,0102	16,47
33,18	1,6	0,3471	0,0105	19,55
36,34	2,0	0,3888	0,0107	22,66
39,24	2,4	0,4305	0,0110	25,75
42,03	2,8	0,4724	0,0112	28,89
44,57	3,2	0,5143	0,0115	31,96
46,95	3,6	0,5563	0,0119	35,01
49,14	4,0	0,5985	0,0122	38,01
51,14	4,4	0,6408	0,0125	40,94
52,95	4,8	0,6831	0,0129	43,77
54,71	5,2	0,7255	0,0133	46,60
56,38	5,6	0,7679	0,0136	49,41
58,19	6,0	0,8100	0,0139	52,37
60,13	6,4	0,8521	0,0142	55,51
62,00	6,8	0,8940	0,0144	58,62
63,86	7,2	0,9355	0,0147	61,76
65,69	7,6	0,9768	0,0149	64,92
67,16	8,0	1,0174	0,0152	67,74
62,10	8,4	1,0513	0,0169	63,67
56,29	8,8	1,0807	0,0192	58,51
48,25	9,2	1,1015	0,0228	50,64
38,46	10,0	1,1342	0,0295	40,96
35,30	10,4	1,1491	0,0326	37,84
31,97	10,8	1,1617	0,0363	34,46
29,39	11,2	1,1746	0,0400	31,85
26,85	11,6	1,1834	0,0441	29,20
24,14	12,0	1,1860	0,0491	26,28
21,20	12,4	1,1772	0,0555	23,00
18,31	12,8	1,1590	0,0633	19,71

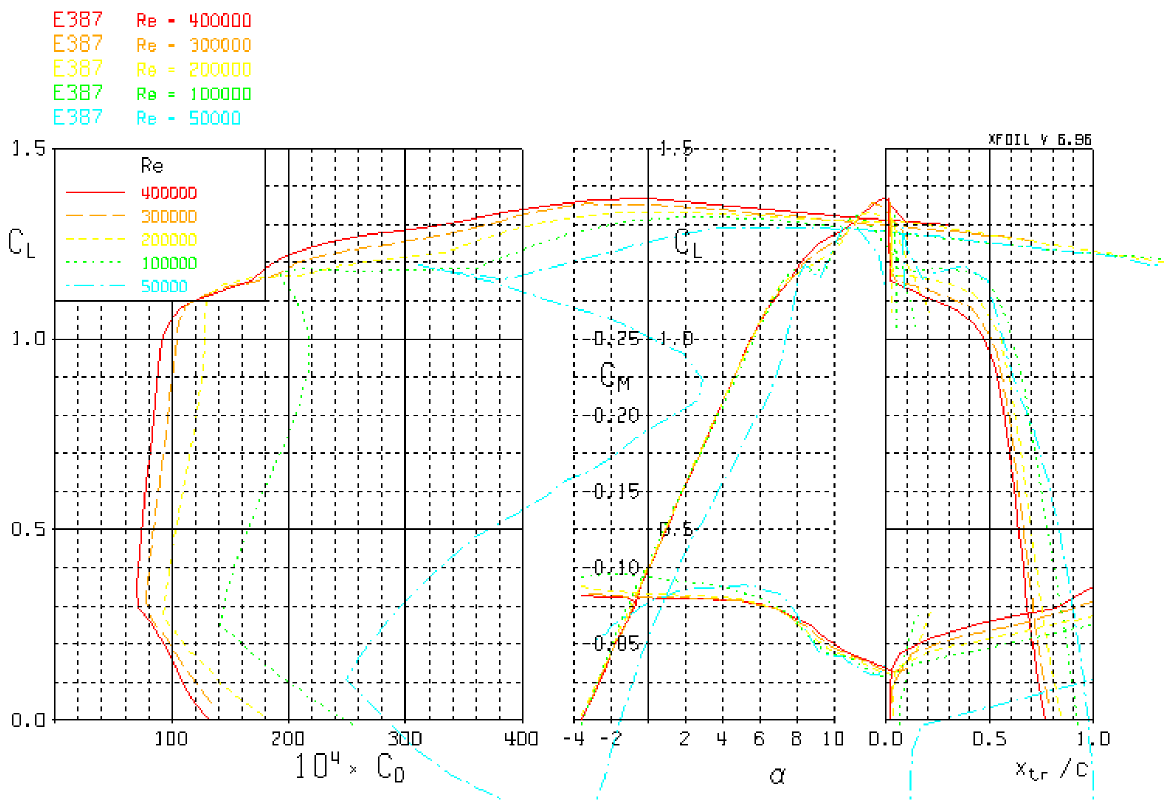


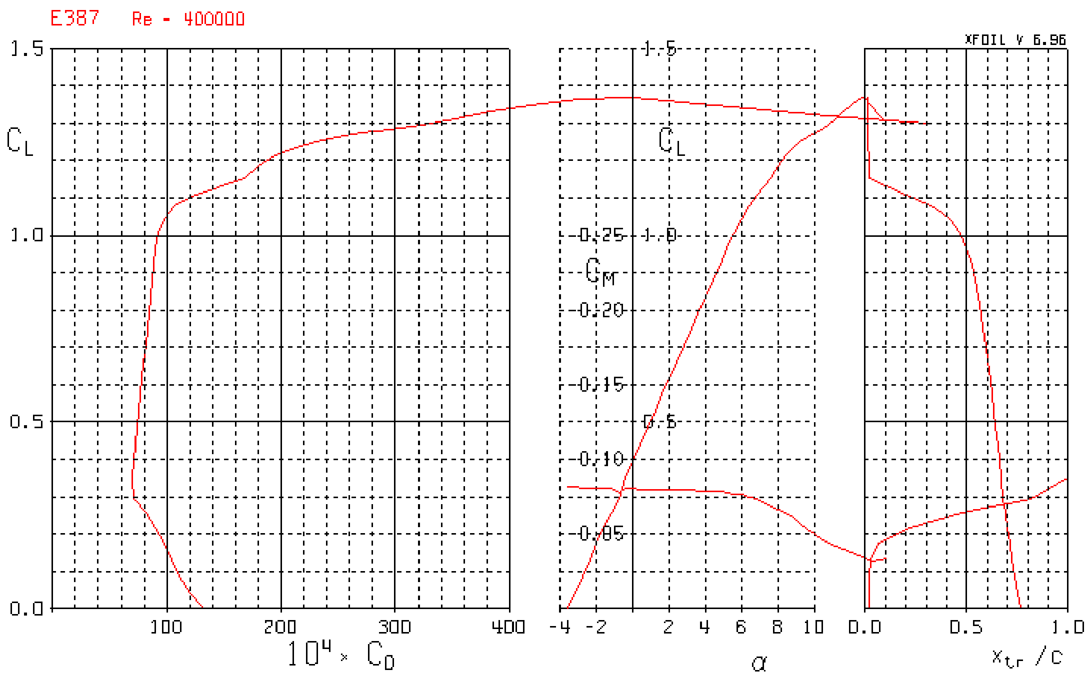
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
17,67	0,4	0,2416	0,0137	8,69
20,09	0,8	0,2821	0,0140	10,67
22,36	1,2	0,3225	0,0144	12,70
24,46	1,6	0,3630	0,0148	14,74
26,41	2,0	0,4038	0,0153	16,78
28,21	2,4	0,4448	0,0158	18,81
29,82	2,8	0,4861	0,0163	20,79
31,28	3,2	0,5274	0,0169	22,72
32,56	3,6	0,5688	0,0175	24,56
33,71	4,0	0,6101	0,0181	26,33
34,75	4,4	0,6515	0,0188	28,05
35,53	4,8	0,6928	0,0195	29,57
36,26	5,2	0,7340	0,0202	31,07
37,09	5,6	0,7751	0,0209	32,65
37,76	6,0	0,8159	0,0216	34,10
38,49	6,4	0,8521	0,0221	35,53
39,65	6,8	0,8965	0,0226	37,54
41,26	7,2	0,9367	0,0227	39,94
43,07	7,6	0,9763	0,0227	42,55
45,50	8,0	1,0152	0,0223	45,85
48,23	8,4	1,0519	0,0218	49,47
47,06	8,8	1,0790	0,0229	48,88
39,54	9,2	1,0981	0,0278	41,44
34,77	9,6	1,1132	0,0320	36,68
30,69	10,0	1,1362	0,0370	32,71
28,91	10,4	1,1539	0,0399	31,05
26,46	10,8	1,1699	0,0442	28,62
23,57	11,2	1,1809	0,0501	25,61
20,96	11,6	1,1883	0,0567	22,85
18,80	12,0	1,1717	0,0623	20,35
16,70	12,4	1,1507	0,0689	17,92
14,75	12,8	1,1289	0,0766	15,67



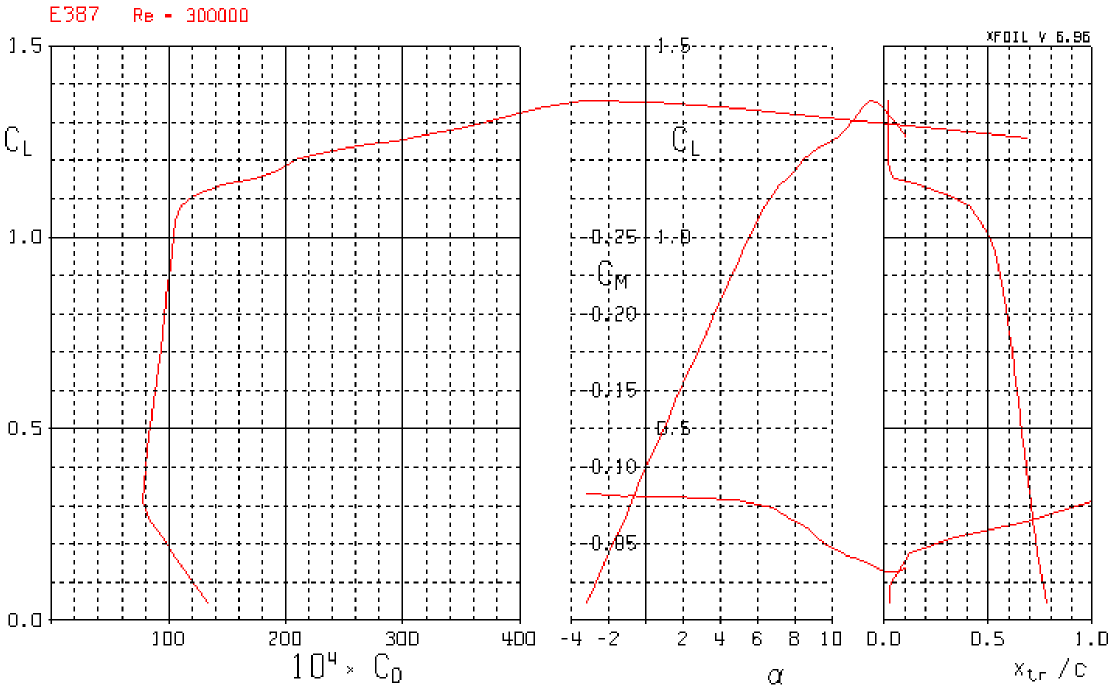
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
17,13	1,2	0,3826	0,0223	10,59
17,94	1,6	0,4190	0,0234	11,62
18,58	2,0	0,4558	0,0245	12,54
19,15	2,4	0,4927	0,0257	13,44
19,61	2,8	0,5296	0,0270	14,27
19,79	3,2	0,5663	0,0286	14,90
19,87	3,6	0,6025	0,0303	15,42
19,83	4,0	0,6381	0,0322	15,84
19,63	4,4	0,6725	0,0343	16,10
19,30	4,8	0,7057	0,0366	16,22
18,86	5,2	0,7373	0,0391	16,19
18,34	5,6	0,7673	0,0418	16,07
17,92	6,0	0,7970	0,0445	16,00
17,15	6,4	0,8212	0,0479	15,54
16,90	6,8	0,8498	0,0503	15,58
16,54	7,2	0,8750	0,0529	15,48
16,73	7,6	0,9057	0,0541	15,92
18,02	8,0	0,9478	0,0526	17,54
20,91	8,4	1,0003	0,0478	20,92
22,58	8,8	1,0412	0,0461	23,04
23,76	9,2	1,0721	0,0451	24,60
24,32	9,6	1,1012	0,0453	25,52
23,34	10,0	1,1268	0,0483	24,77
22,40	10,4	1,1432	0,0510	23,95
18,96	10,8	1,1415	0,0602	20,26
17,52	11,2	1,1347	0,0648	18,67
15,82	11,6	1,1335	0,0717	16,84

12.1.7 **Airfoil ID: *Eppler 387***

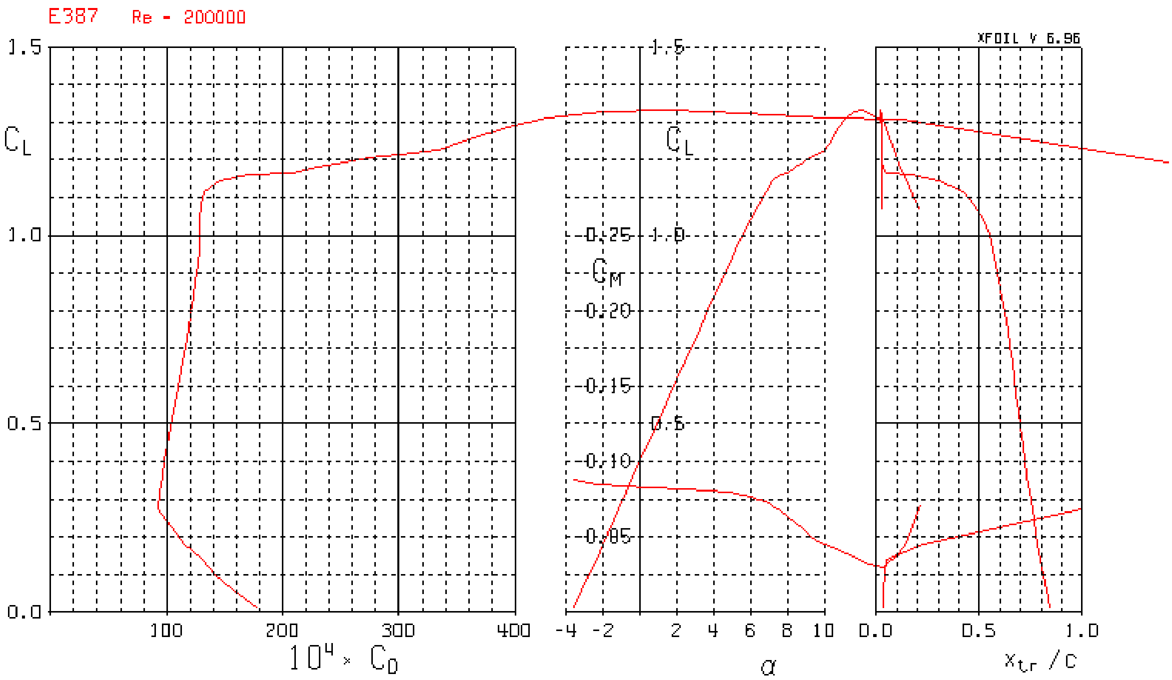




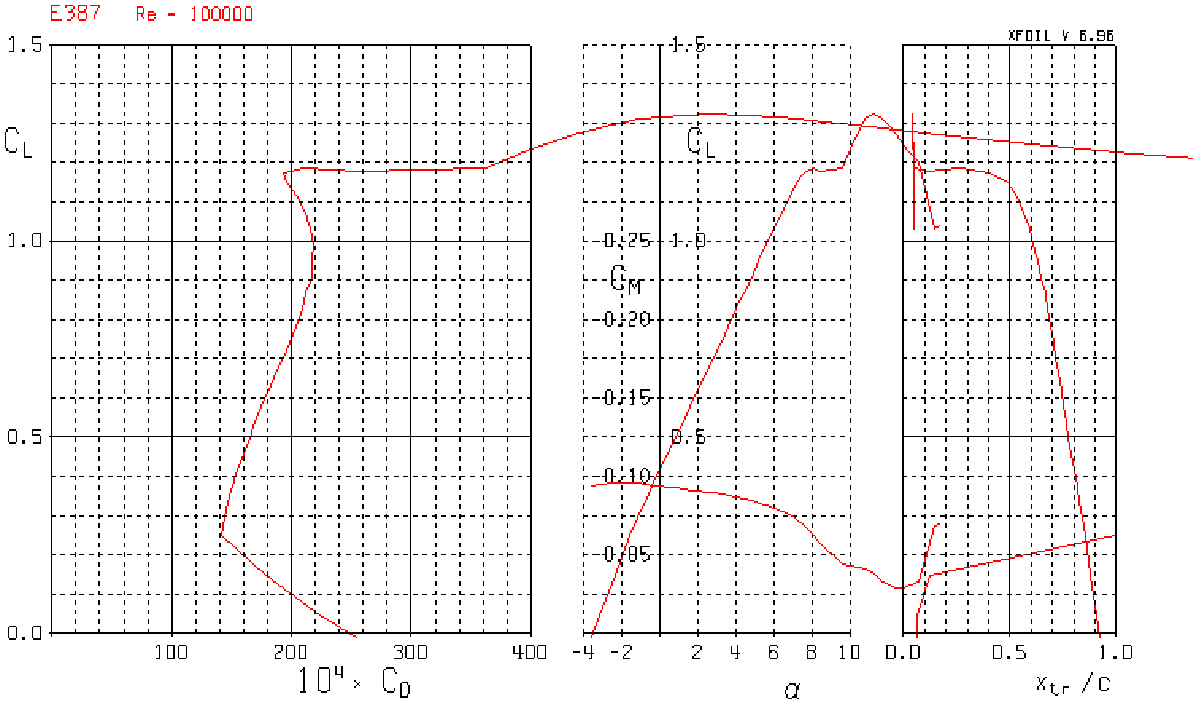
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,06	-3,6	0,0008	0,0132	0,00
3,59	-3,2	0,0433	0,0121	0,75
7,79	-2,8	0,0867	0,0111	2,29
12,54	-2,4	0,1304	0,0104	4,53
17,96	-2,0	0,1746	0,0097	7,51
24,23	-1,6	0,2178	0,0090	11,31
31,79	-1,2	0,2597	0,0082	16,20
41,03	-0,8	0,2938	0,0072	22,24
50,95	-0,4	0,3531	0,0069	30,28
56,22	0,0	0,3969	0,0071	35,42
61,25	0,4	0,4410	0,0072	40,67
66,00	0,8	0,4851	0,0074	45,97
70,67	1,2	0,5293	0,0075	51,41
75,07	1,6	0,5735	0,0076	56,85
79,18	2,0	0,6176	0,0078	62,23
83,13	2,4	0,6617	0,0080	67,62
86,90	2,8	0,7056	0,0081	72,99
90,60	3,2	0,7493	0,0083	78,43
94,28	3,6	0,7929	0,0084	83,95
97,70	4,0	0,8363	0,0086	89,34
100,98	4,4	0,8795	0,0087	94,70
103,75	4,8	0,9223	0,0089	99,63
106,67	5,2	0,9643	0,0090	104,75
108,33	5,6	1,0053	0,0093	108,62
107,37	6,0	1,0447	0,0097	109,74
100,55	6,4	1,0789	0,0107	104,44
88,40	6,8	1,1068	0,0125	93,00
68,77	7,6	1,1546	0,0168	73,89
65,67	8,0	1,1853	0,0181	71,49
62,56	8,4	1,2137	0,0194	68,92
57,91	8,8	1,2335	0,0213	64,32
54,39	9,2	1,2510	0,0230	60,84
50,83	9,6	1,2631	0,0249	57,13
47,19	10,0	1,2742	0,0270	53,27
43,47	10,4	1,2844	0,0296	49,26
40,15	10,8	1,2968	0,0323	45,72
37,27	11,2	1,3119	0,0352	42,69
34,78	11,6	1,3309	0,0383	40,12
32,24	12,0	1,3504	0,0419	37,47
29,67	12,4	1,3635	0,0460	34,65
26,94	12,8	1,3676	0,0508	31,50
22,60	13,2	1,3428	0,0594	26,19
19,46	13,6	1,3201	0,0678	22,36
16,92	14,0	1,2977	0,0767	19,28



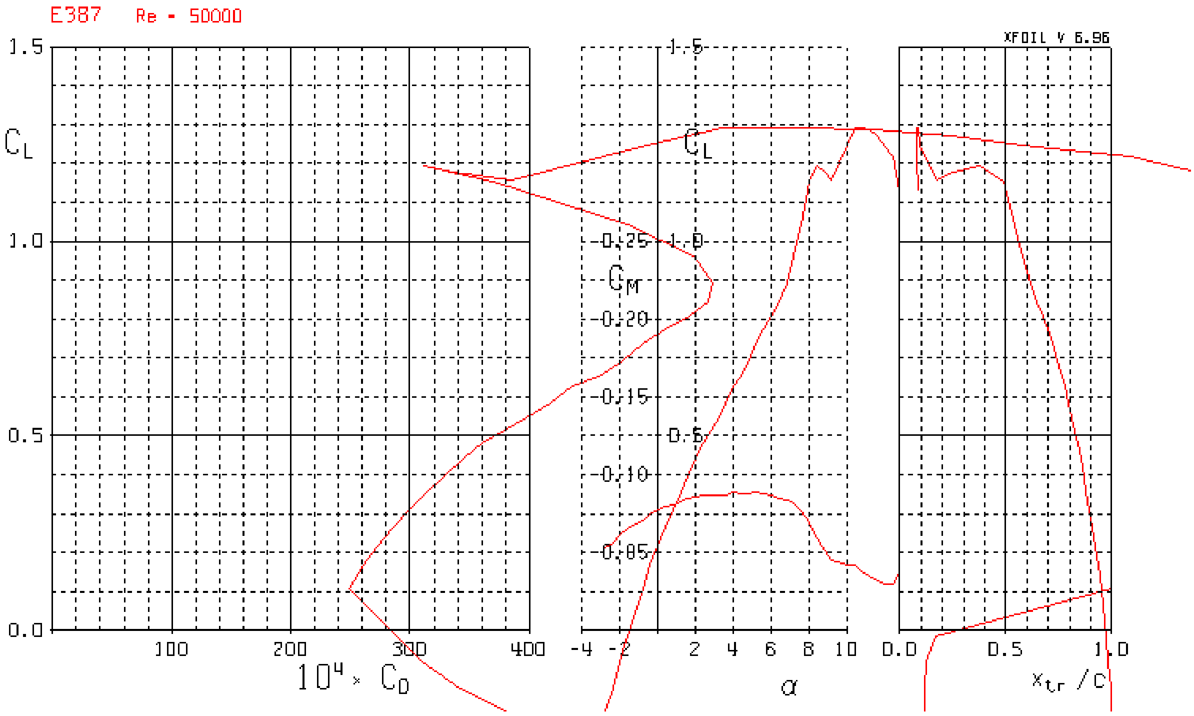
C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
3,53	-3,2	0,0471	0,0133	0,77
7,36	-2,8	0,0900	0,0122	2,21
17,15	-2,0	0,1766	0,0103	7,21
23,09	-1,6	0,2187	0,0095	10,80
30,49	-1,2	0,2558	0,0084	15,42
40,31	-0,8	0,3116	0,0077	22,50
45,11	-0,4	0,3550	0,0079	26,88
49,65	0,0	0,3987	0,0080	31,35
54,04	0,4	0,4426	0,0082	35,95
58,12	0,8	0,4865	0,0084	40,54
62,05	1,2	0,5305	0,0086	45,19
65,81	1,6	0,5745	0,0087	49,88
69,33	2,0	0,6184	0,0089	54,52
72,62	2,4	0,6623	0,0091	59,10
75,84	2,8	0,7061	0,0093	63,73
78,99	3,2	0,7496	0,0095	68,39
82,17	3,6	0,7929	0,0097	73,16
85,23	4,0	0,8361	0,0098	77,93
88,16	4,4	0,8790	0,0100	82,66
90,90	4,8	0,9217	0,0101	87,27
93,75	5,2	0,9638	0,0103	92,04
96,67	5,6	1,0044	0,0104	96,88
98,31	6,0	1,0441	0,0106	100,46
97,38	6,4	1,0819	0,0111	101,29
89,59	6,8	1,1118	0,0124	94,46
78,07	7,2	1,1352	0,0145	83,18
66,73	7,6	1,1524	0,0173	71,63
60,62	8,0	1,1743	0,0194	65,70
57,75	8,4	1,2013	0,0208	63,30
53,14	8,8	1,2174	0,0229	58,63
49,72	9,2	1,2311	0,0248	55,17
46,02	9,6	1,2411	0,0270	51,27
42,61	10,0	1,2528	0,0294	47,70
39,66	10,4	1,2665	0,0319	44,64
36,90	10,8	1,2828	0,0348	41,80
34,39	11,2	1,3090	0,0381	39,35
32,19	11,6	1,3369	0,0415	37,22
29,80	12,0	1,3551	0,0455	34,68
26,38	12,4	1,3518	0,0513	30,67
22,72	12,8	1,3346	0,0587	26,25
19,87	13,2	1,3135	0,0661	22,78
17,31	13,6	1,2884	0,0744	19,65
15,11	14,0	1,2607	0,0834	16,96



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
0,81	-3,6	0,0144	0,0177	0,10
3,60	-3,2	0,0567	0,0157	0,86
6,95	-2,8	0,0981	0,0141	2,18
10,76	-2,4	0,1404	0,0131	4,03
15,81	-2,0	0,1818	0,0115	6,74
29,66	-1,2	0,2744	0,0093	15,54
33,63	-0,8	0,3171	0,0094	18,94
37,43	-0,4	0,3601	0,0096	22,46
41,03	0,0	0,4033	0,0098	26,05
44,45	0,4	0,4467	0,0101	29,71
47,68	0,8	0,4902	0,0103	33,39
50,73	1,2	0,5337	0,0105	37,06
53,53	1,6	0,5771	0,0108	40,67
56,16	2,0	0,6206	0,0111	44,24
58,61	2,4	0,6640	0,0113	47,76
60,97	2,8	0,7073	0,0116	51,28
63,27	3,2	0,7504	0,0119	54,81
65,62	3,6	0,7934	0,0121	58,45
68,03	4,0	0,8361	0,0123	62,21
70,47	4,4	0,8788	0,0125	66,06
72,89	4,8	0,9213	0,0126	69,96
75,39	5,2	0,9627	0,0128	73,97
77,97	5,6	1,0043	0,0129	78,14
81,11	6,0	1,0439	0,0129	82,87
83,89	6,4	1,0814	0,0129	87,24
84,23	6,8	1,1177	0,0133	89,05
78,83	7,2	1,1446	0,0145	84,34
68,14	7,6	1,1611	0,0170	73,42
56,19	8,0	1,1660	0,0208	60,68
51,40	8,4	1,1833	0,0230	55,92
47,33	8,8	1,1956	0,0253	51,75
44,02	9,2	1,2065	0,0274	48,35
40,32	9,6	1,2143	0,0301	44,43
36,85	10,0	1,2283	0,0333	40,84
34,79	10,4	1,2560	0,0361	38,99
32,77	10,8	1,2878	0,0393	37,19
30,68	11,2	1,3114	0,0428	35,13
27,92	11,6	1,3301	0,0476	32,20
25,02	12,0	1,3329	0,0533	28,88
22,22	12,4	1,3247	0,0596	25,58
19,63	12,8	1,3111	0,0668	22,47
17,80	13,2	1,3058	0,0734	20,34
12,26	14,0	1,1904	0,0971	13,38
10,44	14,4	1,1498	0,1102	11,19
8,79	14,8	1,1083	0,1261	9,25
7,28	15,2	1,0663	0,1464	7,52



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
2,12	-3,2	0,0474	0,0224	0,46
8,21	-2,4	0,1468	0,0179	3,14
17,72	-1,6	0,2507	0,0142	8,87
20,39	-1,2	0,2936	0,0144	11,05
22,86	-0,8	0,3358	0,0147	13,25
24,89	-0,4	0,3761	0,0151	15,26
26,82	0,0	0,4168	0,0155	17,32
28,64	0,4	0,4579	0,0160	19,38
30,30	0,8	0,4990	0,0165	21,40
31,78	1,2	0,5400	0,0170	23,36
33,14	1,6	0,5810	0,0175	25,26
34,34	2,0	0,6219	0,0181	27,08
35,42	2,4	0,6627	0,0187	28,83
36,40	2,8	0,7033	0,0193	30,53
37,36	3,2	0,7438	0,0199	32,22
38,39	3,6	0,7846	0,0204	34,00
39,57	4,0	0,8258	0,0209	35,96
41,06	4,4	0,8680	0,0211	38,25
41,77	4,8	0,9060	0,0217	39,76
43,75	5,2	0,9486	0,0217	42,62
45,38	5,6	0,9884	0,0218	45,12
47,86	6,0	1,0300	0,0215	48,58
50,69	6,4	1,0710	0,0211	52,45
54,35	6,8	1,1126	0,0205	57,33
58,77	7,2	1,1478	0,0195	62,96
60,95	7,6	1,1745	0,0193	66,05
56,21	8,0	1,1866	0,0211	61,23
46,03	8,4	1,1769	0,0256	49,93
32,88	9,6	1,1873	0,0361	35,83
30,88	10,0	1,2347	0,0400	34,32
29,01	10,4	1,2753	0,0440	32,76
26,68	10,8	1,3100	0,0491	30,54
24,09	11,2	1,3221	0,0549	27,70
21,59	11,6	1,3139	0,0609	24,74
19,29	12,0	1,2948	0,0671	21,95
17,17	12,4	1,2709	0,0740	19,35
15,23	12,8	1,2452	0,0817	17,00
13,51	13,2	1,2208	0,0904	14,93
12,09	13,6	1,2006	0,0993	13,25
7,07	14,4	1,0314	0,1458	7,18
6,81	14,8	1,0396	0,1528	6,94



C_L/C_D	α	C_L	C_D	$(C_L^{3/2})/C_D$
4,33	-0,8	0,1075	0,0249	1,42
6,38	-0,4	0,1664	0,0261	2,60
7,99	0,0	0,2180	0,0273	3,73
9,24	0,4	0,2639	0,0286	4,75
10,33	0,8	0,3089	0,0299	5,74
11,28	1,2	0,3534	0,0313	6,71
12,09	1,6	0,3968	0,0328	7,61
12,78	2,0	0,4399	0,0344	8,48
13,40	2,4	0,4832	0,0361	9,31
13,59	2,8	0,5151	0,0379	9,76
13,74	3,2	0,5469	0,0398	10,16
13,98	3,6	0,5835	0,0417	10,68
14,41	4,0	0,6283	0,0436	11,42
14,24	4,4	0,6526	0,0458	11,50
14,51	4,8	0,6935	0,0478	12,08
14,85	5,2	0,7370	0,0496	12,75
15,09	5,6	0,7759	0,0514	13,29
15,14	6,0	0,8079	0,0534	13,61
15,30	6,4	0,8422	0,0551	14,04
16,13	6,8	0,8940	0,0554	15,25
17,85	7,2	0,9598	0,0538	17,49
21,63	7,6	1,0439	0,0483	22,10
31,17	8,0	1,1526	0,0370	33,46
38,29	8,4	1,1930	0,0312	41,82
34,96	8,8	1,1772	0,0337	37,93
29,98	9,2	1,1572	0,0386	32,25
25,01	10,0	1,2472	0,0499	27,93
22,95	10,4	1,2905	0,0562	26,08
20,79	10,8	1,2931	0,0622	23,64
18,71	11,2	1,2881	0,0689	21,23
16,83	11,6	1,2696	0,0754	18,97
15,09	12,0	1,2438	0,0824	16,83
13,48	12,4	1,2188	0,0904	14,88
10,93	12,8	1,1273	0,1032	11,60

12.3 Phoenix - geometry and coordinates

Top surface X-axis	Top surface Y-axis	Bottom surface X-axis	Bottom surface Y-axis
1,00000	0,00091	0,00100	-0,00259
0,99000	0,00146	0,00250	-0,00373
0,98000	0,00203	0,00500	-0,00471
0,97000	0,00261	0,00750	-0,00532
0,95000	0,00388	0,01000	-0,00576
0,92500	0,00571	0,01250	-0,00610
0,90000	0,00779	0,01500	-0,00639
0,87500	0,01005	0,02000	-0,00685
0,85000	0,01245	0,02500	-0,00721
0,82500	0,01498	0,03500	-0,00777
0,80000	0,01760	0,05000	-0,00835
0,77500	0,02029	0,07500	-0,00907
0,75000	0,02304	0,10000	-0,00971
0,72500	0,02586	0,12500	-0,01032
0,70000	0,02872	0,15000	-0,01089
0,67500	0,03163	0,17500	-0,01143
0,65000	0,03458	0,20000	-0,01195
0,60000	0,04051	0,22500	-0,01243
0,55000	0,04639	0,25000	-0,01288
0,50000	0,05205	0,27500	-0,01331
0,45000	0,05733	0,30000	-0,01371
0,40000	0,06200	0,35000	-0,01443
0,35000	0,06577	0,40000	-0,01506
0,30000	0,06814	0,45000	-0,01561
0,27500	0,06863	0,50000	-0,01607
0,25000	0,06855	0,55000	-0,01641
0,22500	0,06781	0,60000	-0,01654
0,20000	0,06631	0,65000	-0,01638
0,17500	0,06396	0,67500	-0,01617
0,15000	0,06068	0,70000	-0,01585
0,12500	0,05638	0,72500	-0,01542
0,10000	0,05093	0,75000	-0,01487
0,07500	0,04415	0,77500	-0,01420
0,05000	0,03557	0,80000	-0,01340
0,02500	0,02402	0,82500	-0,01247
0,02000	0,02109	0,85000	-0,01139
0,01500	0,01779	0,87500	-0,01017
0,01250	0,01595	0,90000	-0,00880
0,01000	0,01395	0,92500	-0,00726
0,00750	0,01174	0,95000	-0,00554
0,00500	0,00921	0,97000	-0,00402
0,00250	0,00611	0,98000	-0,00323
0,00100	0,00362	0,99000	-0,00242
0,00000	0,00010	1,00000	-0,00161