

XFLR5

Analysis of foils and wings operating at low Reynolds numbers

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1 Purpose

This document is not intended as a formal help manual, but rather as an aid in using XFLR5. Its purpose is to explain the methods used in the calculations, and to provide assistance for the less intuitive aspects of the software.

2 Introduction

2.1 XFLR5's development history

The primary purposes for the development of XFLR5 were to provide:

- A user-friendly interface for XFOIL
- A translation of the original FORTRAN source code to the C/C++ language, for all developers who might have a need for it

This was done in accordance, and in the spirit of Mark Drela's and Harold Youngren's highly valuable work, which they have been kind enough to provide free of use under the General Public License.

The resulting software is not intended as a professional product, and thus it does not offer any guarantees of robustness, accuracy or product support. It is merely a personal use application, developed as a hobby, and provided under GPL rules for use by all.

For this reason, it should be noted and understood that XFLR5 may not be default-free. Some significant bugs affecting result precision have been reported in the beta releases and corrected.

However, XFLR5 has been thoroughly tested against other software and published experimental results, up to now with some success, and this permits a limited amount of trust in the results it provides.

The algorithms for foil analysis implemented in XFLR5 are exactly the same as those of the original XFOIL code, except for the translation from FORTRAN to C++. No changes nor amendments have been made. The translation in itself could have caused new bugs. However, the code has been thoroughly tested against numerous original XFOIL analyses, always with consistent results. It may be found, in some cases, that one of the two programs may not converge where the other will, or that the path to convergence is different from one to the other. This is due to the different manner in which floating point numbers and calculations are processed by the two compilers. Having said this, the converged results are always close, and any differences within the convergence criteria set in the XFOIL source code.

Hence, both XFOIL and XFLR5 results of airfoil analysis will be referred to herein as "XFOIL results".

Wing analysis capabilities have been added from version 1.00. Initially, this was done at the suggestion of Matthieu Scherrer, who has experimented with his Matlab "Miarex" code the application of the Non-linear Lifting Line Theory (herein referred to as "LLT") to the design of wings operating at low Reynolds numbers.

Later on, the necessity arose to add the Vortex Lattice Method (herein referred to as "VLM") for the design and analysis of wings with geometries not consistent with the limitations of the LLT.

The latest v3.00 version introduces Katz and Plotkin's recommended VLM method based on quadrilateral rings, and the the VLM calculation of planes with elevator and fin.

2.2 Code structure

Five different "Applications" have been implemented:

- Two direct design modes which are convenient to compare foils, and to design new foils with the use of B-Splines
- The mixed inverse (QDES) and the full inverse (MDES) foil design routines, virtually unchanged from the original
- The foil direct analysis routines (OPER)
- The wing and plane design

3 Foil Analysis and Design Modes

3.1 General

This part of the code is built around XFoil and its main features, i.e. the design routines, and the direct and inverse analysis (OPER, MDES, GDES, and QDES). Except for the implementation of the Windows interface, no special feature has been added to these modules.

To run and use XFLR5, no special knowledge nor any previous experience of XFoil is necessary, although users accustomed to XFoil should have no difficulty in recognizing the new Windows-style menu options.

Since the analysis engine is very much unchanged from the original, users are advised to refer to the original XFoil help to understand the purpose, operation, and limitations of the foil direct and inverse analysis. Their use in XFLR5 is basically the same, with a limited number of necessary adaptations for the Windows interface.

3.2 Direct Analysis [Oper]

3.2.1 Foil object

3.2.1.1 Foil Database

Foils are loaded from standard foil files and are stored in a runtime database. Any number of foils may be loaded at any time.

3.2.1.2 File format

XFLR5 recognizes only the plain traditional format for foils, i.e. files which contain the foil's name on the first line, followed by the X,Y coordinates, which run from the trailing edge, round the leading edge, back to the trailing edge in either direction:

```
Foil Name
X (1)  Y (1)
X (2)  Y (2)
.      .
.      .
X (N)  Y (N)
```

All lines containing a '#' character are ignored.

No special checks are performed on the input geometry. Users are advised to check the file format if the foil is not read properly by XFLR5.

3.2.2 Foil Modification

XFLR5 provides the same options for foil modification as the original XFoil code. These are:

- local and global refinement
- modification of the thickness, camber, max thickness and max camber positions.

The modification of these parameters will cause a new foil to be generated.

Whenever a foil is modified, deleted or overwritten, all its associated results are deleted to ensure consistency.

Experience shows, and XFoil advises, that refinement of the foil's panels, after it has been loaded or modified, is usually a prudent measure to take before any analysis.

3.2.3 Analysis/Polar object

Unlike XFoil, an analysis of a given foil may be performed only after a 'polar object' has been defined and associated to this foil. The results of the analysis will automatically be associated and added to the polar object.

Any number of polars may be created and associated to a given foil.

A polar object is defined by:

- its Type
- its Reynolds and Mach numbers
- laminar to turbulent transition criterion
- forced trip locations on top and bottom surfaces

By default, the transition number is set to 9, and the trip locations are set at the trailing edge.

In addition to the Type 1, 2 and 3 polars which are unchanged from XFoil, Type 4 polars have been introduced, showing data for a given angle of attack at variable Re. The purpose is to enable determination of the critical Re value.

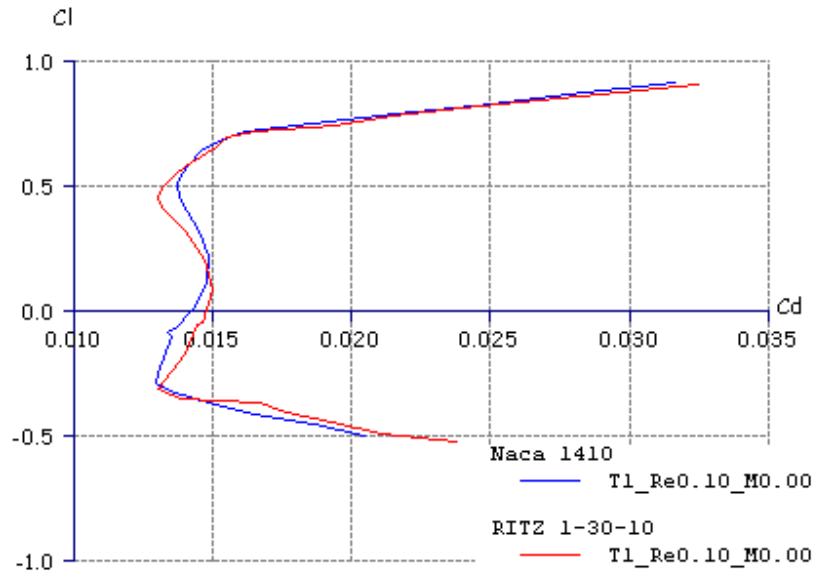


Figure 1 : Type 1 foil polars

3.2.4 Operating Point (OpPoint) object

An operating point of a given foil is defined by its angle of attack and its Re number. Always associated to a foil and to a Polar object, the OpPoint stores the inviscid and viscous results of the analysis.

Any number of OpPoints may be stored in the runtime database, the only limitation being computer memory. OpPoints may use significant memory resources.

To insure consistency, any modification to the foil or to the polar causes the operating point to be deleted from the database.

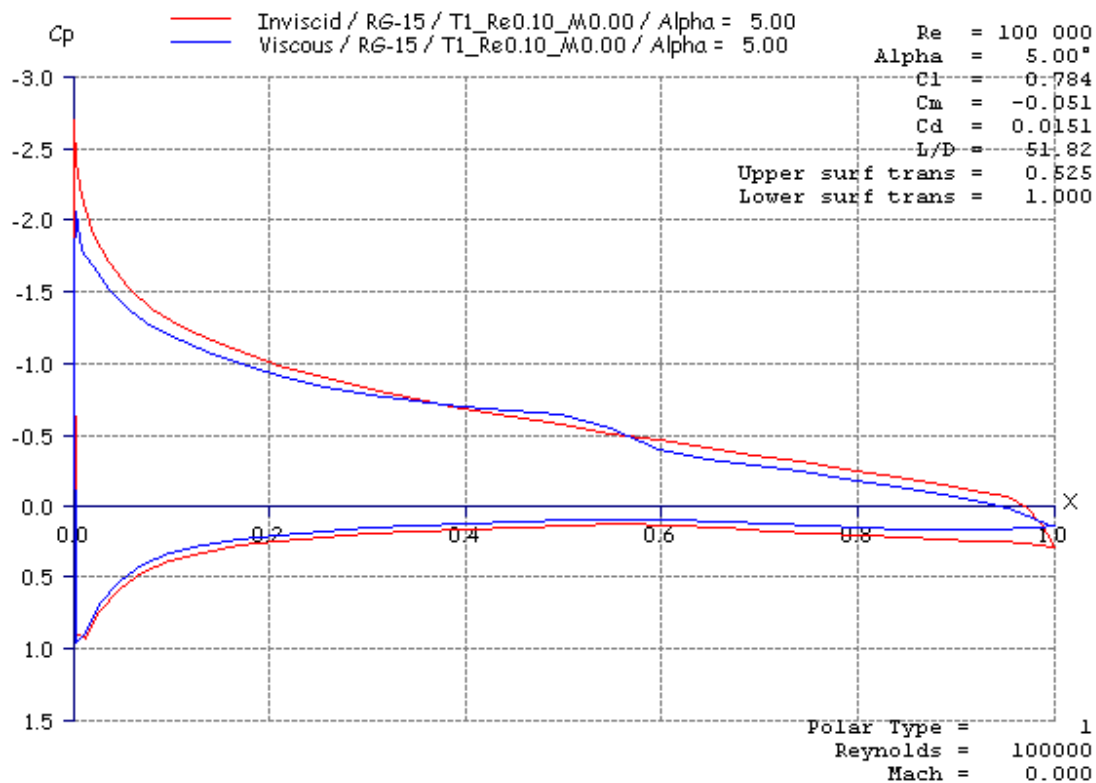


Figure 2 : C_p calculation

3.2.5 XFOil analysis

Each time an XFOil direct analysis is performed and the convergence is achieved, an OpPoint is generated and the values of interest are stored in the currently selected polar object. Data is added to the polar, whether the option to store OpPoints has been activated or not.

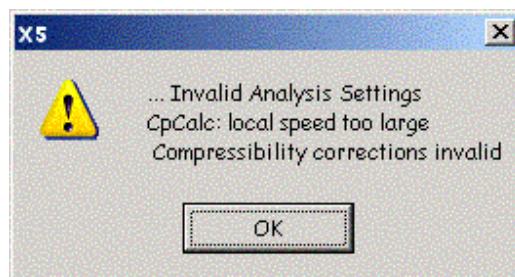
An XFOil calculation performed at the same angle of attack and Re as an existing OpPoint causes the latter to be replaced, and the polar data to be updated.

The "Init BL" checkbox is the equivalent of the "Init" menu command in XFOil, i.e., it resets the boundary layer to standard values before an analysis. It is recommended to check the box at the time of the first calculation, and whenever the analysis of an OpPoint is unconverged or is very different from the previous one.

In the case of sequential analysis, the "Init BL" is automatically deactivated after a first converged point has been reached, and is reset after an unconverged calculation.

3.2.6 XFOil errors

Given the complexity and difficulty of a viscous analysis, XFOil is remarkably robust and consistent. It may happen however that the following error message is generated during an analysis.



This error message is usually caused by a too coarse paneling of the foil, or a too sharp leading edge. It is possible that in such a case XFoil gets "stuck" and fails at any attempt to perform a new analysis. The menu command "Operating Point/Reset XFoil" can be used to reinitialize all the variables and reset the currently selected foils and polars.

3.2.7 Session example – Direct Analysis

1. Load a foil from a file
2. [Optional] Use the "Refine Locally" or "Refine Globally" commands to optimize the foil's panels
3. Use the "Define Analysis/Polar" command in the Polar menu, or F6, to define an analysis – for instance a Type 1 analysis at $Re = 100,000$ and $Mach = 0.0$
4. Define an angle of attack or a lift coefficient to analyze – for instance $\alpha = 0^\circ$
5. Click on the "Analyze" button in the right toolbar to launch an analysis
6. If the XFoil analysis has converged, the C_p distribution will be automatically displayed
7. Check the Show BL or Show Pressure buttons to visualize either distribution
8. Check the "Sequence" button in the right toolbar
9. Define the min and max angles for the analysis – for instance from $\alpha = -6^\circ$ to $\alpha = 10^\circ$
10. Since the new start value is significantly different from the last calculation (i.e. $\alpha = 0^\circ$), check the "Init BLs" button
11. Click on the "Analyze" button
12. Click on the "Animate" button to visualize modifications of the boundary layer or pressure distributions with angle of attack variations
13. Click on the "Polars" command in the View menu, or type F8
14. Use the mouse button and wheel to drag and zoom the graphs

3.3 Full Inverse Design [MDES] and Mixed Inverse Design [QDES]

3.3.1 General

Both design modes are unchanged from the original.

Foils generated by the Full Inverse method are defined by 255 coordinate points, which is excessive for subsequent Direct Analysis. A re-paneling of the foil is strongly recommended.

Although foils generated by the Mixed Inverse Method have the same number of panels as the original foil, a re-panel is still advisable.

3.3.2 Session example – Full Inverse Design

1. Switch to the Full Inverse Application (Menu command or Ctrl+3)
2. Select a foil from the loaded database, or load a foil from a file
3. Click on the "New Spline" button in the right toolbar dialog
4. Select two points either on the upper or lower surface, but not one on each
5. Drag the spline's control points to define a new speed distribution
6. Click on the "Apply" button to register the change
7. Click on the "Execute" button to calculate the new foil geometry
8. Use the mouse buttons and wheel to drag and zoom the graph and the foil
9. To store the modified foil, click on the arrow in the top toolbar, or select "Store foil in the database" in the Foil menu
10. Switch to the Direct Analysis Application (Menu or Ctrl+5)
11. Use "Refine globally" in the Design menu to generate a coarser panel
12. Proceed with the direct analysis

3.3.3 Session example – Mixed Inverse Design

Steps 1 to 6 are identical to the Full Inverse design method

7. Click on the "Mark for Modification" button to define which part of the foil is to be modified
8. Click on the "Execute" button to calculate the new foil geometry
9. Check for convergence in the text window
In the case of non-convergence, it is possible either to resume iterations by clicking again the "Execute" button, or to export the modified geometry as it is

Finish as with the Full Inverse design method.

4 Foil Design

4.1 General

A crude design module has been included in XFLR5, which allows the design of Foils either from B-Splines or from "Splined Points". The former gives smoother surfaces, the latter authorizes greater control over the geometry.

This design mode however is not the best way to design foils, and the other possibilities derived from XFOil are far more adapted and recommended i.e.:

- modification of a foil's thickness and camber
- interpolation of foils
- inverse methods

This foil design mode, however, is useful to overlay different foils and compare their geometries.

4.2 B-splines main features

Upper and lower surfaces are each determined by a separate and single B-Spline.

Spline degree can be chosen between 2 and 5.

4.3 Spline Points main features

Upper and lower surfaces are each determined by a set of control points

Control points are linked by 3rd degree B-Splines

Two intermediate control points are added to the link splines, at 1/3 and 2/3 respectively of the separation of the two control points. These two points are added automatically and are not visible, nor can they be modified

The slope at each visible control point is determined by the line passing through the immediately previous and next control points

4.4 Leading and trailing edge

For both methods, the slope at the leading edge is vertical and may not be changed. In the case of a design from B-Splines, this is done by forcing the second control point to remain on the vertical axis.

In the case of "Splined Points" the slope at the trailing edge is determined by the position of two supplementary rear points, one for each surface.

4.5 Output precision

The maximum number of output points on each surface is 150. This is consistent with the sizing of the XFOil arrays, and with the precision required for the application, although the increase of computing power and memory capacity of modern computers could allow for more points. Typically, XFOil requires at least 50 points on each side to perform an adequate analysis.

In both cases, it is prudent to "re-panel" the foil in the main menu, to improve the convergence of the XFOil analysis and its precision. This can be done with the equivalent of XFOil's "PANE" and "CADD" commands. Upon exit from the design module, the user is asked whether to export or not the foil to the analysis module.

4.6 Reference foils

The "Reference Foils" are essentially provided as display guidelines. However, they can be also be modified from within the Direct Design applications with the XFOil routines LERAD, TGAP, TFAC, TSET etc.

5 Wing Analysis

5.1 Theory - General

XFOil provides unique insight in the behavior of airfoils, but is a 2D analysis, hence the results are those of a wing of infinite aspect ratio and which is defined with a single airfoil. The influence that the aspect ratio alone may have on the wing's polars, let alone the sweep or the dihedral, justifies the need for a more sophisticated wing analysis.

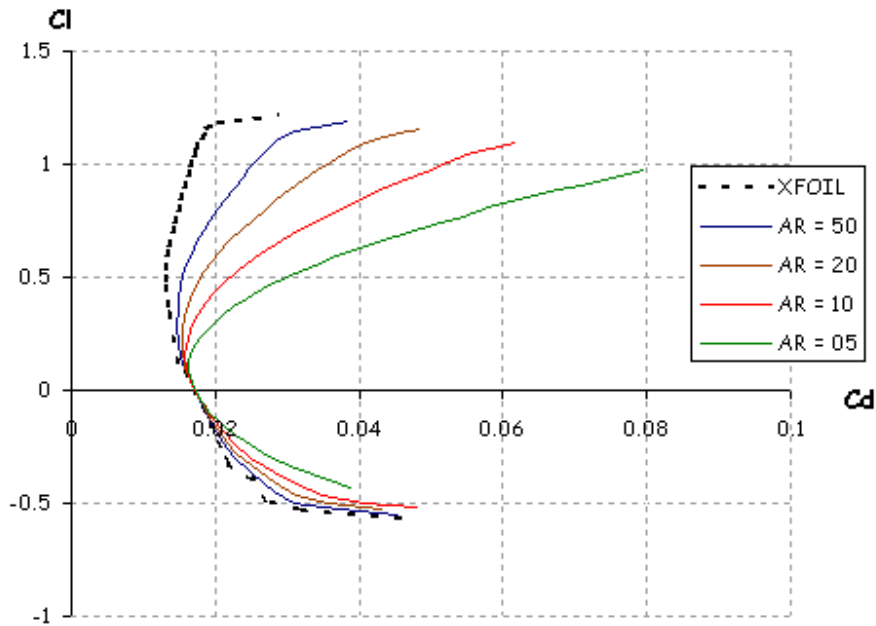


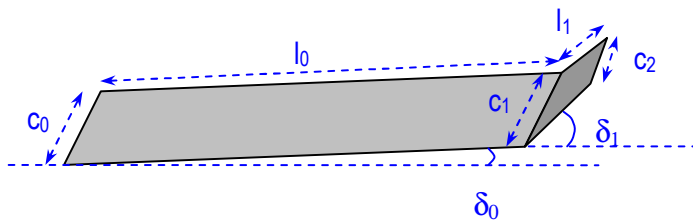
Figure 3 : Influence of Aspect Ratio - LLT Calculation NACA 3412 Airfoil - Taper Ratio = 1 - Sweep = 0°

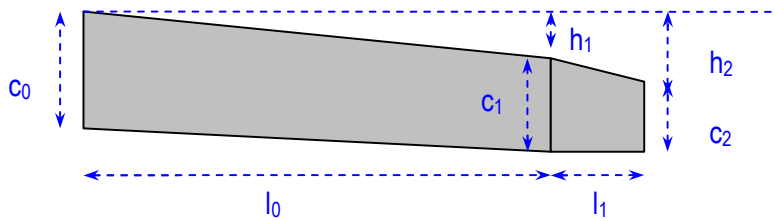
The wing may be computed by either one of two methods, each having its own advantages, and both having some usage restrictions.

The first is a Lifting Line method, derived from Prandtl's wing theory. The second is a Vortex Lattice method.

The originality of both implementations is their coupling with XFOil calculation results to estimate the viscous drag associated with the wing, although this is done in a different manner for each method.

5.2 Wing Definition





The wing is defined as a set of panels. Each panel is defined by :

- its length l_i ,
- the root and tip chords c_i and c_{i+1} ,
- the leading edge offset at root and tip chords h_i and h_{i+1}
- the dihedral angle δ_i
- the mesh for VLM analysis.

The spanwise length of a panel should be at least equal to the minimum length of the VLM elements on other panels. Divisions by zero or non-physical results could result from panels of insufficient length.

Twist (washout) is processed in LLT as a modification of the angle of attack.

In VLM, twist is processed as a change of the geometry.

The 'span' of the wing is defined as

$$S = 2 \times \sum l_i$$

For ease of interpretation, the wing is shown developed on a horizontal planform, both in the wing design dialog box and in the 2D view. Only the 3D view gives a 'realistic' representation of the geometry.

A wing may be asymmetric if the foils are different on each side. This option is meant to provide some capability to evaluate the influence of flaps, but should be used with caution. It has been tested neither against experimental nor theoretical results.

5.3 General Limitations

As a general rule, LLT and VLM are adapted to configurations of thin lifting surfaces, and operating at small angles of attack.

Probably the most questionable assumption of the wing design algorithm is the use of XFOIL transition results to wings with finite aspect ratio. The 2D simulation proposed by XFOIL corresponds to infinite wings, where a laminar bubble extends indefinitely along the span. Some authors suggest that on span-limited wings, such bubbles will appear only on a fraction of the planform. However, theories for 3D transitions are still in development and to the author's knowledge, not giving total satisfaction yet.

The method which consists in interpolating XFOIL generated results is clearly an approximation with no real theoretical or experimental background, but should be a reasonable approximation for wings with moderate to high aspect ratio.

The viscous characteristics will be less and less representative as the wing geometries differ from the ideal 2D Xfoil infinite wing. Hence those results for non planar geometries, low aspect ratio or high sweep should be considered with caution.

5.4 Lifting Line Theory (LLT) - Non Linear

5.4.1 General

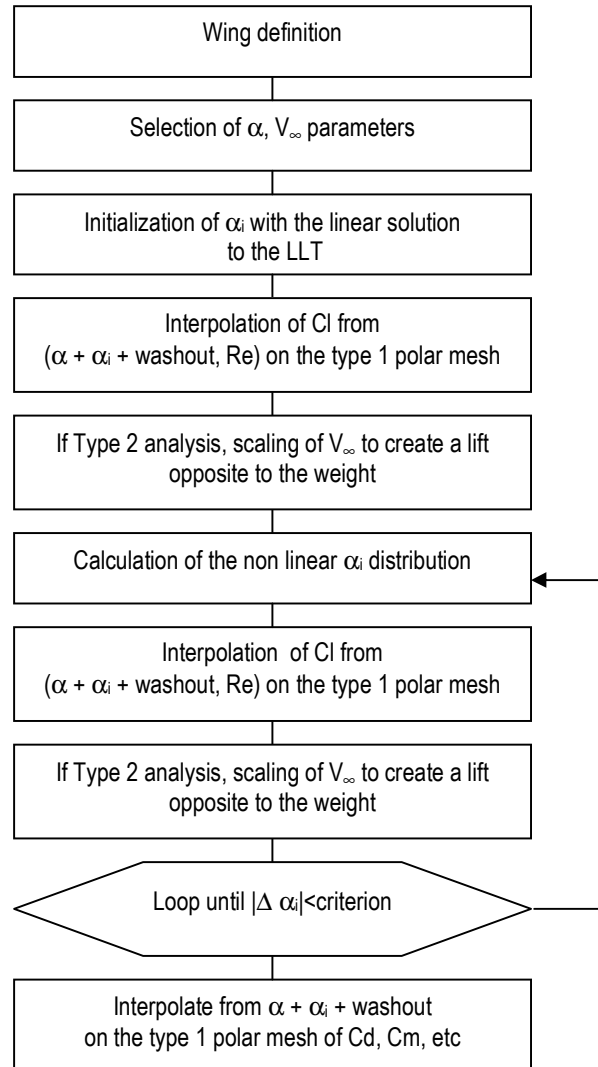
The 'classic' LLT is linear, i.e. the relation $Cl = f(\alpha)$ is linear, and viscous effects are not taken into account. In the present application, a non-linear LLT has been implemented based on the NACA technical note 1269 [1].

Quote from Technical Note 1269:

"The hypothesis upon which the theory is based is that a lifting wing can be replaced by a lifting line and that the incremental vortices shed along the span trail behind the wing in straight lines in the direction of the freestream velocity. The strength of these trailing vortices is proportional to the rate of change of the lift along the span. The trailing vortices induce a velocity normal to the direction of the free-stream velocity. The effective angle of attack of each section of the wing is therefore different from the geometric angle of attack by the amount of the angle (called the induced angle of attack) whose tangent is the ratio of the value of the induced velocity to the value of the freestream velocity. The effective angle of attack is thus related to the lift distribution through the induced angle of attack. In addition, the effective angle of attack is related to the section lift coefficient according to two-dimensional data for the airfoil sections incorporated to the wing. Both relationships must be simultaneously satisfied in the calculation of the lift distribution of the wing.

If the section lift curves are linear, these relationships may be expressed by a single equation which can be solved analytically. In general however, the section lift curves are not linear, particularly at high angles of attack, and analytical solutions are not feasible. The method of calculating the spanwise lift distribution using non-linear section lift data thus becomes one of making successive approximations of the lift distribution until one is found that simultaneously satisfies the aforementioned relationships."

In the present implementation, the non linear lift behavior is interpolated on pre-generated meshes of XFoil Type 1 polars and the non-linearity is solved by an iteration loop :



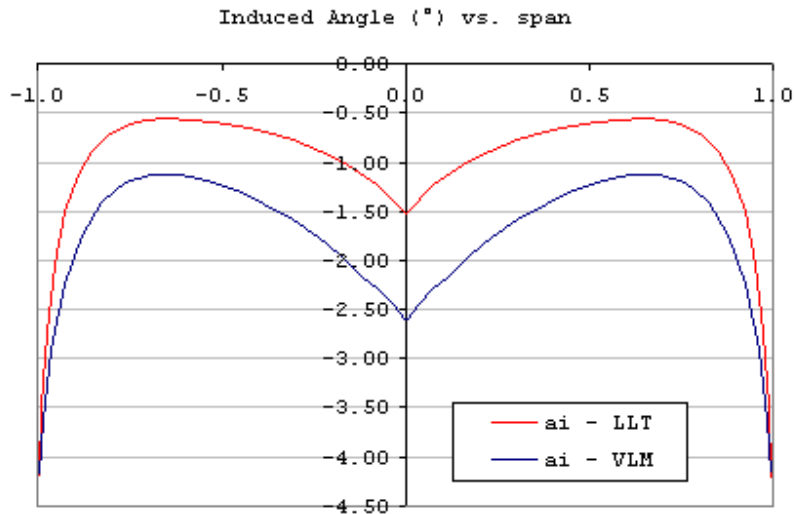


Figure 4 –Induced Angle – Bi-Airfoil NACA3412-NACA1410 – AR = 14.8 – TR = 2.0 – Alfa = 5° - V= 16.7m/s

5.4.2 Limitations of the LLT

It is important to note that the lifting line theory has two main limitations. Quote from Technical Note 1269 :

"The calculations are subject to the limitations of lifting line theory and should not be expected to give accurate results for wings of low aspect ratio and large amounts of sweep"

In addition, the wing's planform is expected to lie essentially in the X-Y plane.

5.4.3 Precautions with the LLT

As it turns out, the convergence of the non-linear LLT is not a robust process, and requires careful use of a relaxation factor. This factor should always be greater than 1. A value of 20 is usually a good start, and may be increased as necessary for convergence.

Usually wings with low aspect ratio require high relaxation value.

The number of stations across the wing span should be chosen around 20, but may be increased up to 40. Greater numbers do not improve the precision of the analysis, but tend to seriously hinder the convergence. The relaxation factor should be increased with higher numbers of span stations.

5.4.4 2D vs. 3D

The LLT assumes implicitly that all the surfaces lie essentially in the X-Y plane.

The only use for the sweep and the dihedral in this implementation of the LLT is for the calculation of the pitching moment coefficient C_m .

Sweep and dihedral are not used in the calculation of the lift distribution.

5.4.5 Lift force

The position of the lift center at each span location is calculated using the usual approximation for thin airfoils, i.e. :

$$X_{CP} = 0.25 - \frac{C_{m0}}{C_l}$$

5.4.6 Downwash

The downwash is defined at each span station as

$$V_i = V_\infty \sin(\alpha_i)$$

For convenience, it is represented at the wing's trailing edge in 3D views.

5.5 Vortex Lattice Method (VLM) - Linear

5.5.1 VLM General principles

A VLM method has been implemented as an alternative, for the analysis of those wing geometries which fall outside the limitations of the LLT.

The main differences from the LLT are:

- The calculation of the lift distribution, the induced angles and the induced drag is inviscid and linear i.e. it is independent of the wing's speed and of the air's viscous characteristics.
- The method is applicable to any usual wing geometry, including those with sweep, low aspect ratio or high dihedral, including winglets.

The principle of a VLM is to assimilate the perturbation generated by the wing to that of a sum of vortices distributed over the wing's planform. The strength of each vortex is calculated to meet the appropriate boundary conditions, i.e. non penetration conditions on the surface of the panels.

A comprehensive description of the principles of VLM analysis is well outside the scope of this document. Only the main features necessary to a sound use of the code are detailed hereafter.

The resolution of the VLM problem requires the inversion of a square matrix of the size of the number of panels. This inversion is performed by Gauss' partial pivot method.

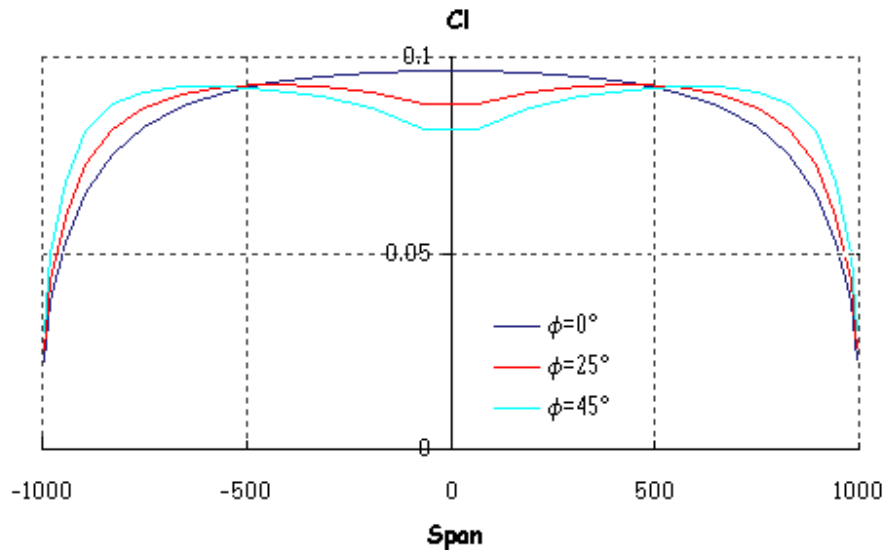


Figure 5 : Influence of Sweep for a given CL - AR=10 - TR=1 - Symmetric Airfoil

5.5.2 VLM Mesh

Ideally, the precision of the calculation increases with the mesh's refinement, but so does the calculation times. It is fairly simple to experiment to determine what is the best compromise for a given design objective.

The mesh should be refined wherever the geometry changes, i.e. at leading and trailing edges, and at panel junctions.

Too fine a mesh in some cases can lead to numerical errors and non-physical results.

5.5.3 Reference surface

The reference area for all wing and plane aerodynamic coefficients is the main wing's area.

5.5.4 Lift force and lift coefficient

The force acting over each panel is the vectorial cross product

$$\mathbf{F} = \rho \mathbf{V} \times \mathbf{\Gamma}$$

$\mathbf{\Gamma}$ being the vortex strength,

ρ is the fluid density

\mathbf{V} is the freestream speed

Which implies that the force is normal to each panel.

The lift coefficient is defined as

$$C_L = \frac{1}{\rho S V^2} \sum_{\text{panels}} \mathbf{F} \cdot \mathbf{n}$$

\mathbf{n} is the normal to each panel

S is the sum of the panels' area, i.e. the planform's area

This formula is applicable either to a chordwise strip or to the wing's total surface.

5.5.5 Induced drag

Up to V2.00, the induced drag has been calculated by integration of surface forces at the 3/4 pt of the VLM panels. After some research in various publications, it seems that this method is less reliable than the integration of the wake kinetic energy in a far field plane, i.e. the Trefftz plane.

From V2.01 onwards, the induced drag is calculated using the latter method.

The pitching moments and centre of pressure position at each span location are calculated by summation of the lift force over the panels.

5.5.6 Linear and non-linear behavior

Traditional VLM analysis do not account for viscous effects. For model aircraft operating at a few m/s however, the viscous drag is not negligible compared to the induced drag, and must therefore be estimated by an alternative mean.

In the present application, the viscous drag is estimated by interpolation of XFOIL pre-generated polars from the C_l value resulting from the linear VLM analysis. This assumes implicitly that the foil's behavior on a finite wing is not very different than on an "infinite XFOIL wing". There is no real background, neither theoretical nor experimental, to support this approach, so it should be used with caution.

As is generally the case when transposing 2D results to 3D analysis, the estimation of viscous drag is probably too low and may lead to arguably optimistic results.

Because the method is linear, it does not, among other things, account properly for stall at high angles of attack, unlike, potentially, the LLT.

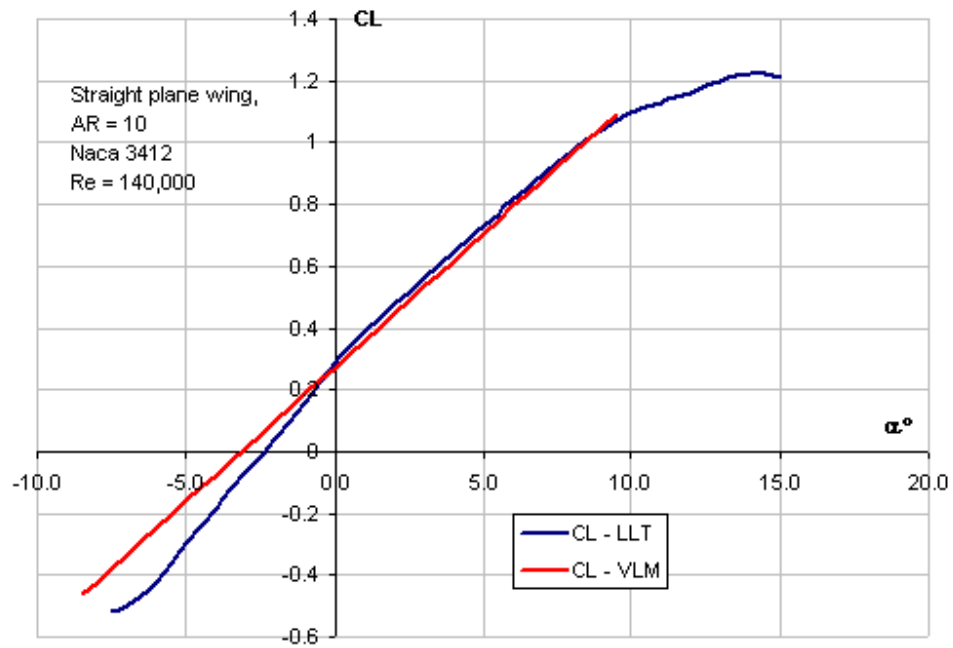
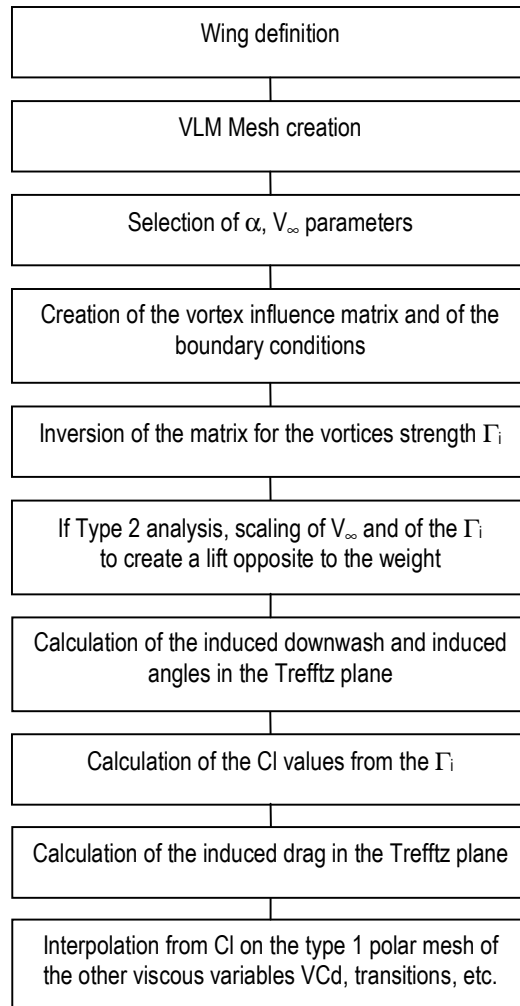


Figure 6 - Linear and Non-Linear modeling

5.5.7 Non-Linear Implementation



5.5.8 Limitations of the VLM

1. Sideslip is not taken into account in the current implementation.
2. The VLM algorithms first computes the lift coefficient C_l and the other values which may be calculated by integration of the surface forces, i.e. the moment coefficients and the center of pressure's position. The other variables (viscous C_d , transitions, etc) are interpolated from the value of C_l on the previously XFOIL-generated polars. This obviously raises an issue for high and low C_l , where the Type 1 polar curve may be interpolated either before or after the stall angle.
VLM results should therefore not be considered around angle of attack values close to stall angles.

5.5.9 Mesh

The wing is "meshed" into a number of panels distributed over the span and the chord of the planform, and a vortex is associated to each panel.

In the current implementation, the panels are placed on the airfoil section's mean thickness line, hence taking into account camber, but not thickness, of the section.

It is recommended to choose a panel distribution which is consistent with the wing's geometry, i.e. the density of the mesh needs to be increased at geometrical breakpoints and at the root and tip of the wings.

A cosine type distribution is recommended in the chordwise direction to provide increased density at the leading and trailing edges.

There is a lower limit size for the panels below which the calculation becomes unstable.

5.5.10 VLM alternative Methods

In the 'classic' VLM method, a horseshoe vortex is positioned at the panel quarter chord and the non-penetration condition is satisfied at the three-quarter chord point.

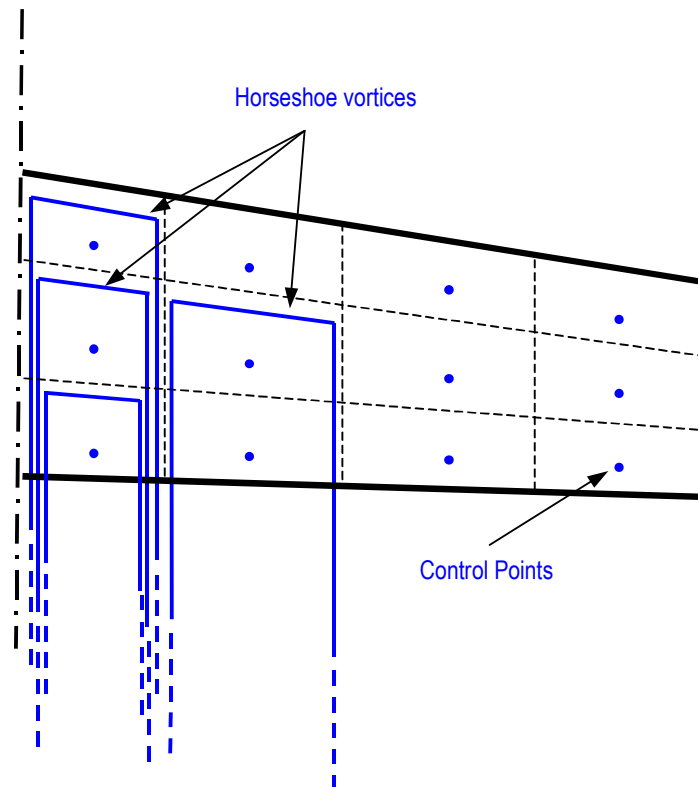


Figure 7 : Classic VLM Method

In the method recommended by Katz and Plotkin [3], only the trailing vortices extend to infinity.

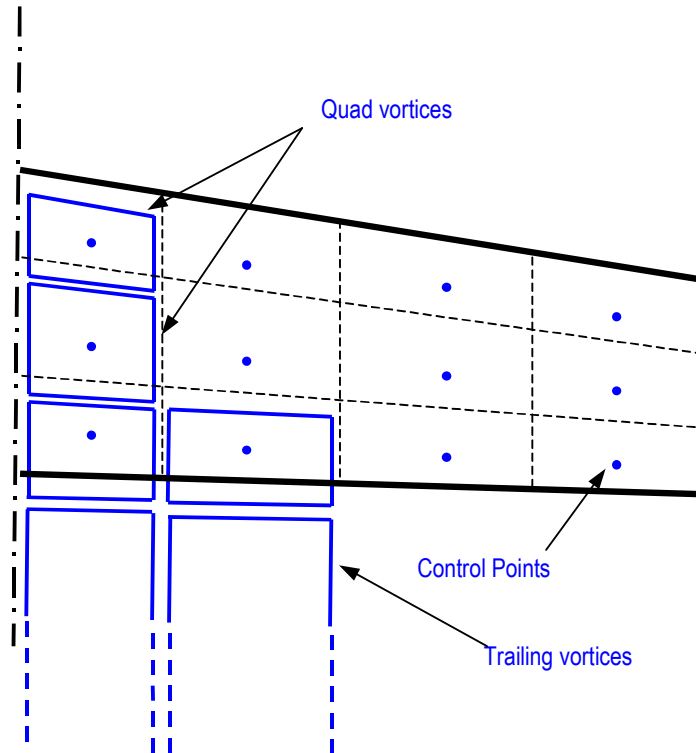


Figure 8 : Quad VLM Method

Since the wake must be force free, the strength of the trailing vortex is equal to that of the trailing edge quad vortex.

Both methods are implemented for comparison, but give close if not identical results in most cases.

5.5.11 Panel disposition

The resolution of the system and the determination of the vortices' strengths require a matrix inversion. In some rare cases, this matrix may turn out to be singular due to a conflicting disposition of panels and control points on the wing's planform.

The problem arises when a control point is positioned on the line of a vortex. This will result in a division by zero. In those cases, manual re-paneling of the wing is sufficient to fix the problem.

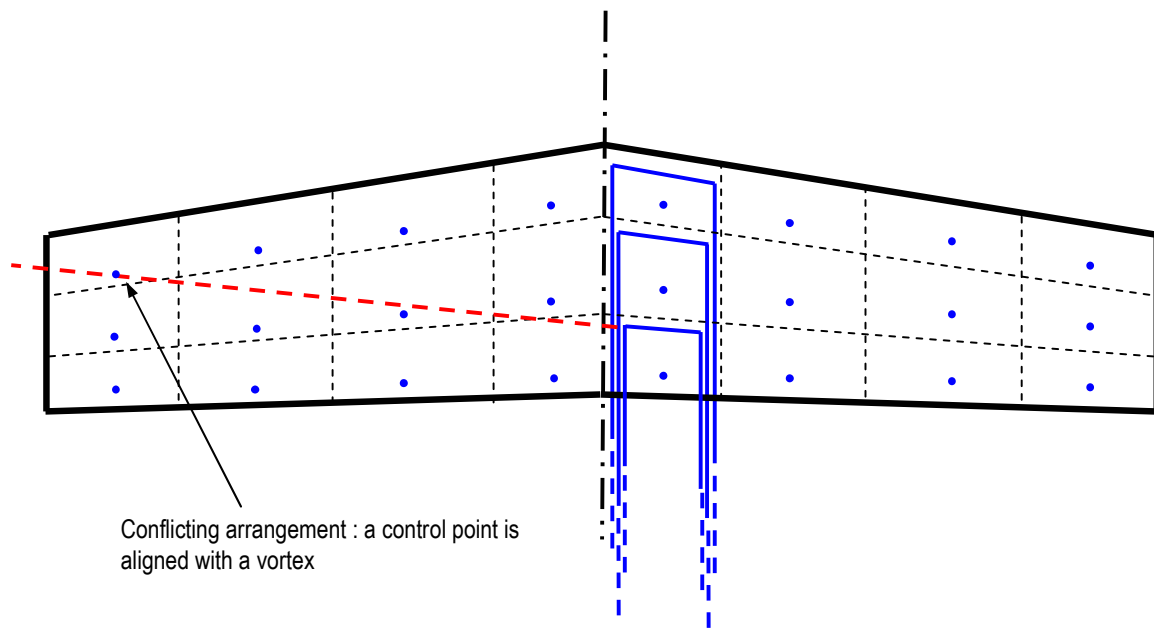


Figure 9 : Quad VLM Method

If inversion problems persist despite re-paneling, it will be necessary to check the consistency of the input data.

5.5.12 Moments

All moment calculations in LLT are strictly in accordance with the formula of NACA TN1269

All moment calculations in VLM are done by integration of surface force applied on the mean camber line.

Moment		Nature	LLT	VLM
Pitching	Airfoil	Moment of lift forces around ¼ chord point	$PCm = \frac{1}{4} - Cl/Cm0$	Sum of moments created by surface forces on the mean camber line
	Viscous	Moment of the viscous airfoil drag forces with respect to XcmRef	Integration of the moment over the wing's lifting line.	Included in the geometric pitching moment, even though it is usually negligible
	Geometric	Moment of lift and induced drag forces at ¼ chord point with respect to XCmRef	Integration of the moment over the wing's planform. Both sweep and dihedral are taken into account	
Rolling		Moment of the lift forces with respect to the plane y=0	Integration of the moment over the wing's lifting line. Dihedral <u>is not</u> taken into account	Integration of the moment over the wing's planform. Dihedral <u>is</u> taken into account $Rm = \sum_{\text{panels}} c_{li} s_i y_i \cos(\delta_i)$
Yawing	Profile	Moment of the viscous airfoil drag forces with respect to the plane y=0	Integration of the moment over the wing's planform.	
	Geometric	Geometric moment induced by the fin's deflection	N/A	Integration of the moment over the wing's planform.
	Induced	Moment of the induced tangential forces with respect to the plane y=0	Integration of the moment over the wing's planform.	

5.5.13 Downwash

From v3.00 onwards, the downwash is calculated at the wing's trailing edge.

5.5.14 Lift coefficient planform distribution

For VLM calculations on the wing's mean camber surface, the lift coefficient of the panel k is:

$$Cp_k = \frac{2\Gamma_k \Delta y_k}{S_k V_\infty}$$

with:

- Γ_k the vortex's strength
- Δy_k the panel's width
- S_k the panel's surface
- V_∞ the freestream speed

5.6 Oswald's factor

Oswald's factor is a measure of the deviations of the wing's induced drag from the optimal elliptic loading and is defined as

$$e = \frac{CL^2}{\pi \cdot AR \cdot ICd}$$

where

CL is the lift coefficient

ICd is the induced drag coefficient

AR is the wing's Aspect Ratio

Oswald's factor should always be smaller than 1. It may happen however that this factor becomes greater than 1 for numerical reasons both in LLT and VLM calculations.

In LLT, this may be corrected by increasing the precision required for convergence, for instance with the following parameters :

- Number of stations = 40
- Relaxation factor = 40
- convergence criterion = 0.001
- Max Number of iterations = 300

5.7 Wing Operating Points and Wing Polars

The presentation of results is the same as for foil analysis, i.e. each converged analysis generates an operating point and adds results to a polar object. The definition and selection of an Analysis/Polar object is necessary to perform a calculation.

Any number of Operating Points may be stored in the runtime database, the only limitation being computer memory. Wing operating points will use significant memory resources.

Type 1 and Type 4 polars are unchanged from foil analysis.

A type 2 polar corresponds to a plane with a given weight operating at constant lift.

For a given angle of attack, the plane's velocity is calculated to create a lift force opposite to the weight of the airplane:

$$V = \sqrt{\frac{2mg}{\rho SC_l \cos(\alpha)}}$$

The angle of descent is

$$\gamma = \arctan\left(\frac{C_d}{C_l}\right)$$

and the horizontal and vertical speeds are respectively

$$\begin{aligned} V_x &= V_\infty \cos(\gamma) \\ V_z &= V_\infty \sin(\gamma) \end{aligned}$$

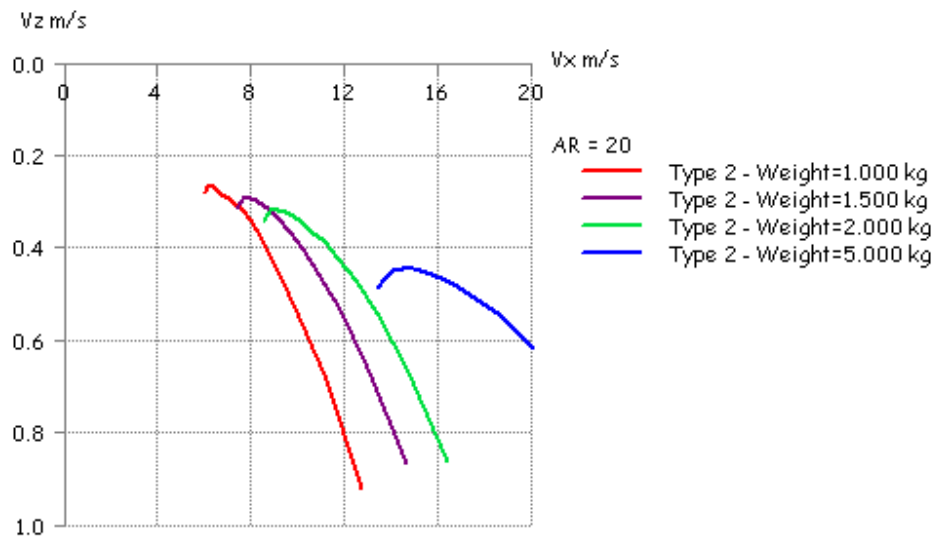


Figure 10: Speed polars based on Type 2 analysis

Convergence for type 2 polars requires that the apparent angle of attack be greater than the zero-lift angle. Otherwise, no speed whatsoever may generate a positive lift...

5.8 Interpolation of the XFOIL-generated Polar Mesh

The code does not recalculate with XFOIL each operating point at every wing station and at each iteration:

- this would require lengthy -and unnecessary- calculations
- XFOIL's convergence is too uncertain

Instead, the operating point is interpolated from a pre-generated set of Type 1 polars.

The wing calculation requires that a set of **Type 1** foil polars be previously loaded or generated for each of the wing's foils.

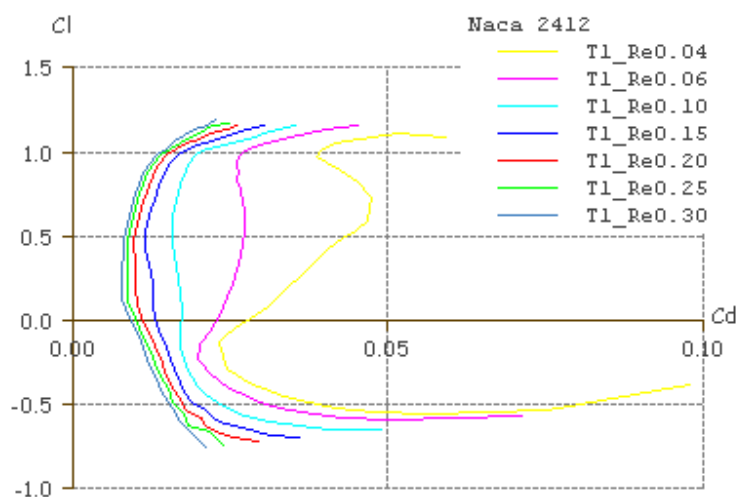


Figure 11: Polar mesh ranging from $Re = 40,000$ to $Re = 300,000$

The set of polars should cover the whole flight envelope of each point of the wing, with regard to both Reynolds numbers and to the apparent angle of attack.

If any point of the wing planform operates outside the available polar mesh, a warning message is issued in the log file. It also hinders the convergence. This happens for instance in the case of short tip chords of elliptic wings. In such a case, the "closest point" of the mesh will be used, and the operating point may be generated and added to the current polar, if so chosen by the user.

For the interpolation process, the code uses without discrimination all available type 1 polars related to the selected foil. The user must therefore be careful to provide only a HOMOGENEOUS AND CONSISTENT set of polars.

The interpolation process of a variable X (X being Cl, Cd, Cm, Transition points etc.) from $[\alpha = \alpha_0 + \alpha_1 + \text{washout}, Re]$ at a geometrical point P between the foils 1 and 2 is:

1. For the first foil, find polars 1 and 2 such that $Re_1 < Re < Re_2$;
if no polar can be found, return on error
if Re is less than all the polars' Reynolds numbers, use the polar of smallest Re
if Re is greater than that of any polars' number, use the polar of greatest Re
2. Interpolate each polar with α to get X_{11} and X_{12} ;
if either polar is not defined up to α , use the smallest or greatest angle available, and issue a warning
if only one polar is available, interpolate only that polar, and issue a warning
3. Interpolate X_1 pro-rata of Re between Re_1 and Re_2
4. Do the same for the second foil to get X_2
5. Interpolate X pro-rata of the position of the point between the two foils

5.9 3D View

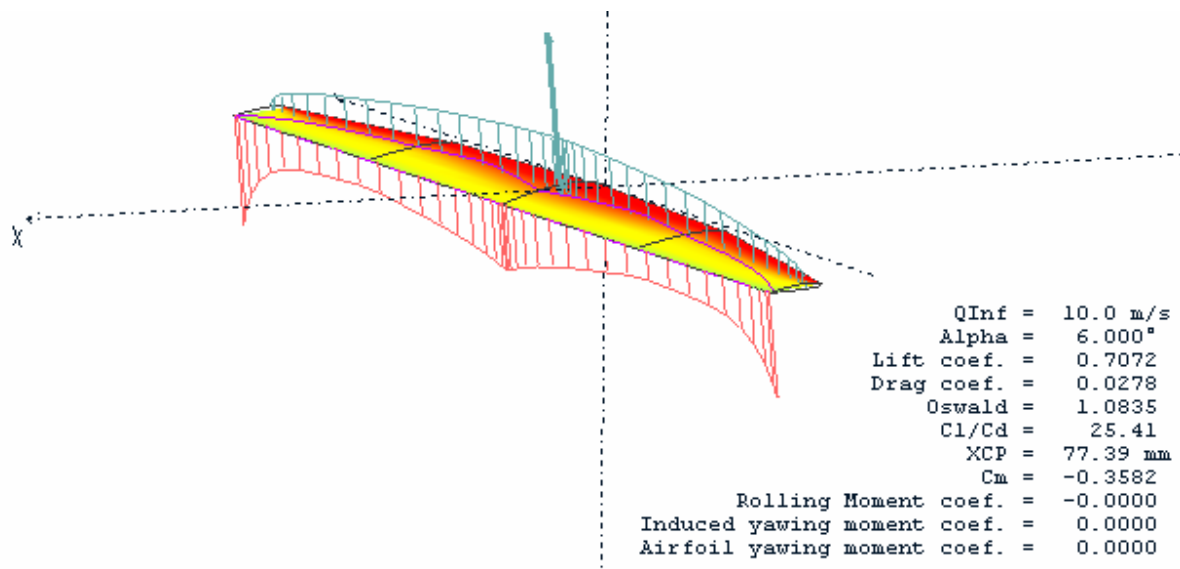


Figure 12 - 3D View

Consistent with all the other views, translation is activated by the mouse left button, zoom by the wheel or middle button, rotation by simultaneous action of the left button and of the "Ctrl" key. The zoom option may not work if the middle button settings are not set to "default" in the Windows interface.

Upwards and lateral mouse movements in the center portion of the screen will cause respectively rotations about the X and Y axis. Horizontal mouse movements on the top and bottom areas of the client window will cause rotation about the model's Z axis.

Display of the C_l and pressure distribution over the wing planform is only available for VLM calculations. LLT, by nature, only calculates a lifting line.

5.10 Choice of the Analysis Method

The LLT method should always be preferred if the wing's geometry is compatible with the limitations of the theory. LLT provides better insight into the viscous drag, gives a better estimation of the behavior around stall conditions at high angle of attack, and is better supported by theoretical published work.

VLM analysis is preferable for all other cases.

5.11 Comparison to Calculated and Experimental Results

The code has been tested against experimental results and against other software, with consistent results. Also, the VLM and LLT algorithms in their XFLR5 implementation are totally independent, but give close results in the linear part of the C_l vs. α plots.

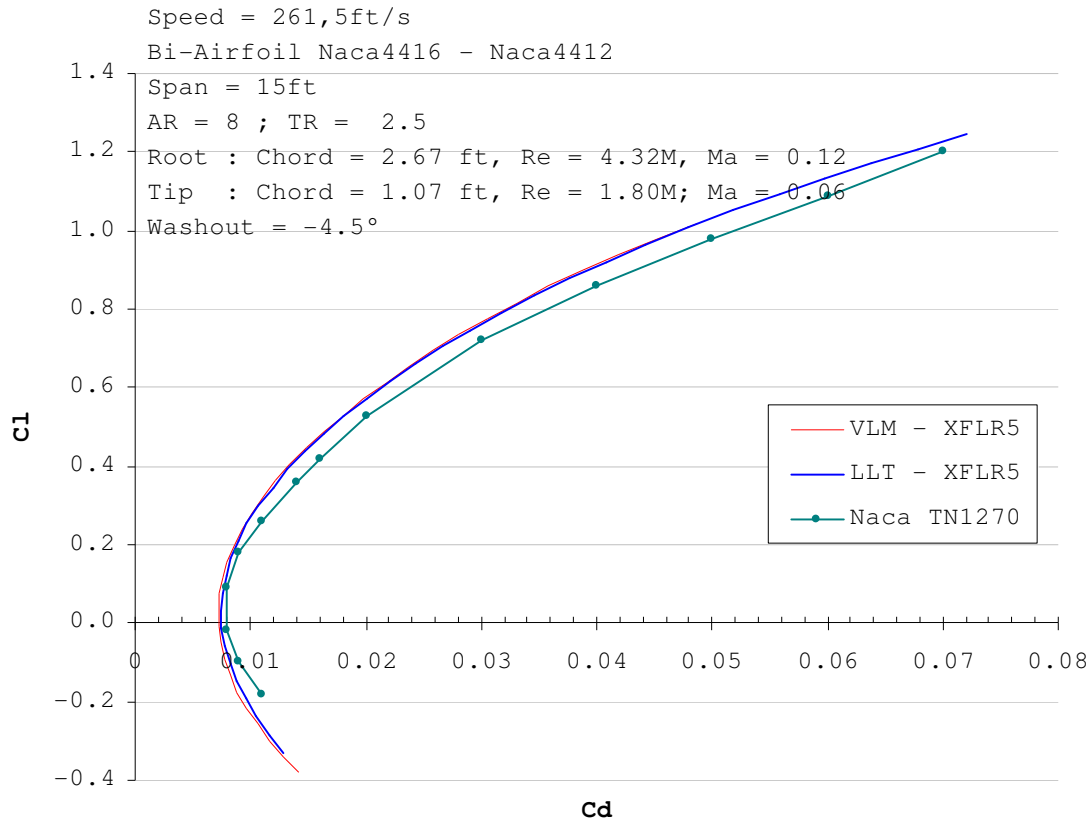


Figure 13: Comparison to test results from Naca Tech. Note 1270

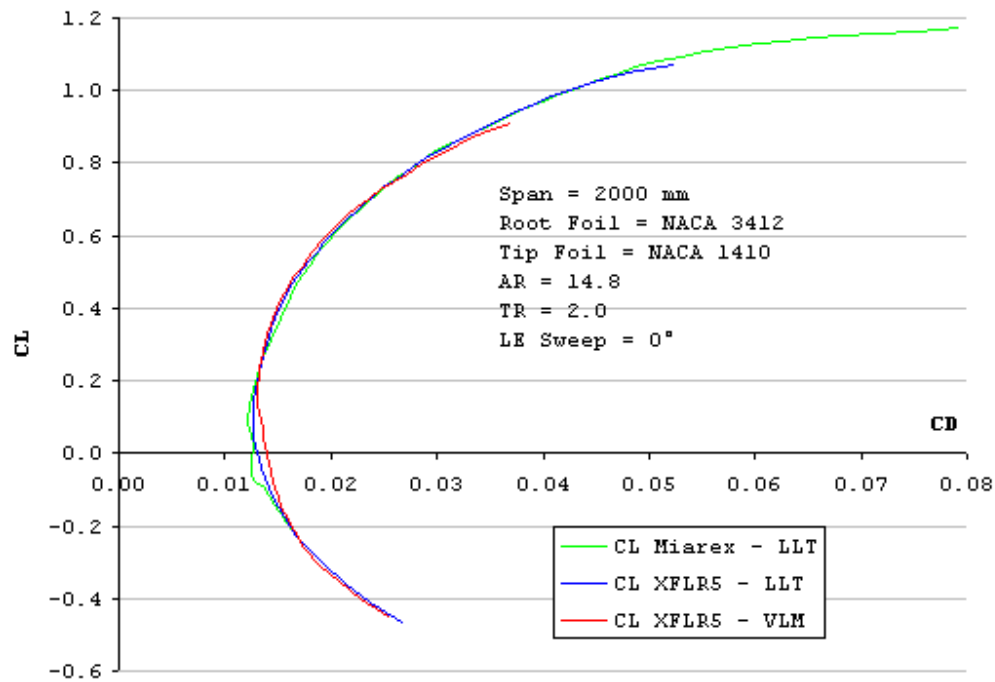


Figure 14: Comparison to results from Miarex

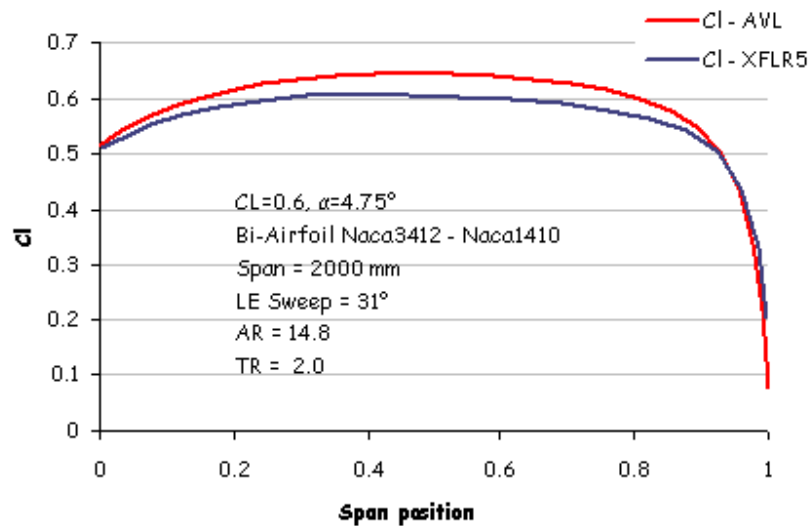


Figure 15: Comparison to AVL

5.12 Session example – Wing Analysis

1. Load the foil(s) which will be used to define the wing
2. In the Direct Analysis Application, click the "Run Batch Analysis" command in the Polars menu, or type Shift+F6
3. Run a batch analysis with the following parameters (make sure these values cover the whole flight envelope of the wing):
 - from $\alpha = -6^\circ$ to $\alpha = 10^\circ$
 - from $Re = 40,000$ to $Re = 160,000$ every 20,000
 - from $Re = 200,000$ to $Re = 500,000$ every 50,000
4. Close the dialog box
5. Use the "Save Associated Polars" in the "Current Foil" menu to save the polars to a ".plr" file for use in future projects
6. Switch to the Wing Design Application or type Ctrl+6
7. Click the "Define Wing" command, or F3 in the Wing menu
8. Define the wing and close the dialog box
9. Click the "Define Analysis/polar" in the Wing Polar menu, or type F6
10. Activate the Type 2 check box
11. Define the plane weight and the center of gravity position (the moment ref. location)
12. Unless the wing has either low aspect ratio, high sweep, or high dihedral, select the "LLT" checkbox, and close the dialog box (↔and ↔)
13. Leave the LLT settings to the default values in the "Operating Point" menu, i.e. "Relax Factor = 20" and "N° of Stations Along the Span = 20"
14. Select an angle of attack in the right toolbar which can be expected to give positive lift equal to the model weight at reasonable Speed/Re values – for instance $\alpha = 3^\circ$
15. Click the "Analyze" button in the right toolbar
16. Click the "3D view" command in the View menu
17. Use the mouse to zoom and rotate the model
18. Use Sequence to calculate a complete wing's polar
15. Click on the "Polars" command in the View menu, or type F8 to visualize the polar graphs

5.13 Non convergences

	Cause	Fix
Both methods	One or more foils cannot be found : the foil(s) may have been renamed or deleted	Check the foils names
	The foils' Type 1 polar meshes do not cover the available flight envelope [most usual case of non convergence]	Extend the foils' Type 1 polar meshes
	In Type 2 analysis, the lift is negative	Calculate only for higher values of the angle of attack
	In Type 2 analysis, the speed is either too low or too high, leading to OpPoints outside of the available flight envelope	Extend the foils' Type 1 polar mesh ; The speed will tend towards infinite values at low aoa, and symmetrically will tend towards 0 at high aoa
	The tip chord is too small and yields too low Reynolds numbers	Either: <ol style="list-style-type: none">1. Check the "Store OpPoints outside the Polar mesh" checkbox2. Omit the end of the wing in the definition of its planform
LLT	The relaxation factor is too small	Increase the factor in the "LLT Settings..." dialog box
	The number of points over the planform is too high	Decrease the number in the "LLT Settings..." dialog box
VLM	The matrix is singular because of a conflicting disposition of VLM panels	Regenerate a manual VLM mesh

The log file will indicate which points of the flight envelope could not be calculated.

The "log file" is a plain text file. If the document does not show up when called from the menu, it may be necessary to manually associate the ".log" extension to Windows' Notepad.

6 Plane Analysis

6.1 General

A plane consists in a main wing, and optionally of an elevator and one or two fins.

The reference area for the calculation of the aerodynamic coefficient is the wing's surface.

The plane calculation uses the same implemented algorithms than for pure wing analysis.

The calculation of a wing with two winglets is equivalent to the calculation of a plane with two fins instead of the winglets. However, the reference area in the second case will not include the fins, hence the aerodynamic coefficients will be slightly greater than in the first case.

6.2 Panel arrangement

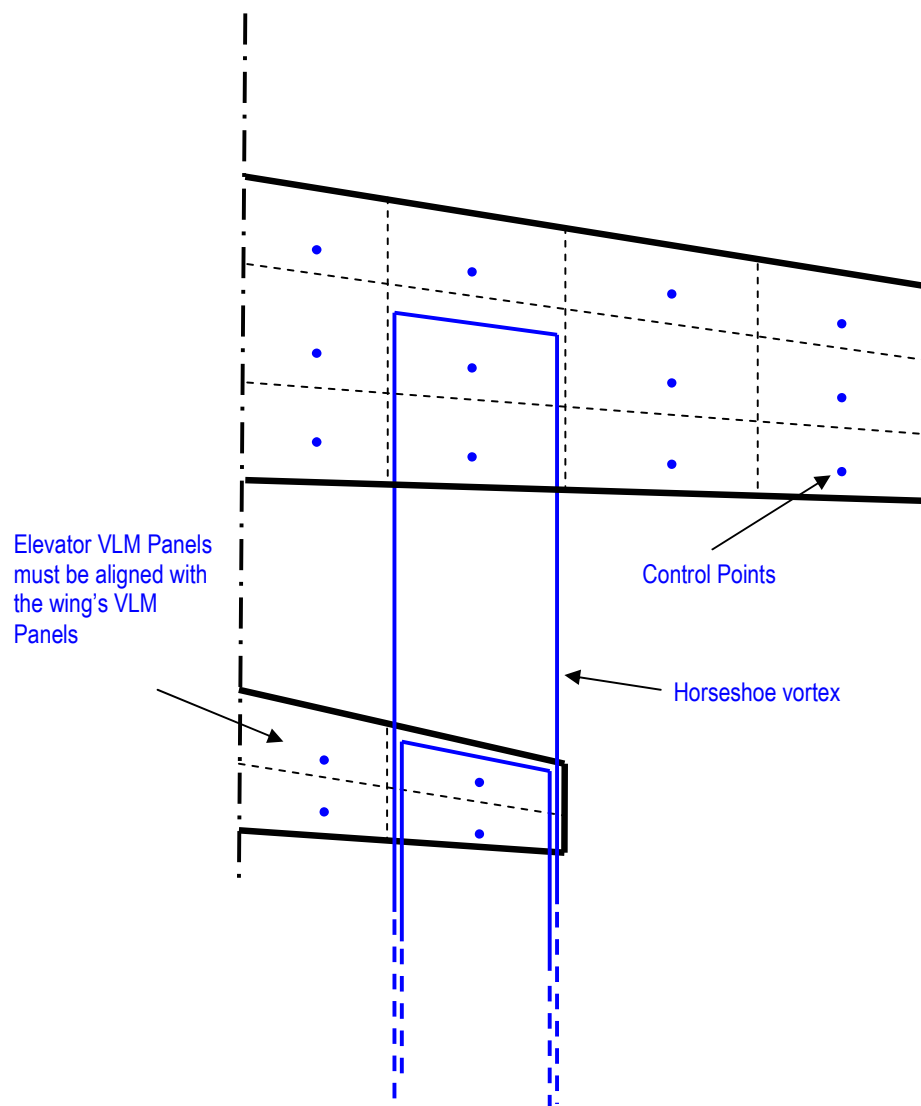


Figure 16: VLM Panel arrangement for a plane

Special care must be taken in the disposition of VLM panels to avoid having a control point of the tail's surface close to the trailing leg of one of the wing's horseshoe vortex. This would lead to a division by zero, and inconsistent results.

The alignment of the panels is done by introducing a section in the wing at the span location of the tail. If this isn't done manually, the code will attempt to introduce manually the section, using the wing's inner foil.

For the same reason, it is a good idea, though not compulsory, to position fins in the planes of the wing's panel junctions.

7 Code Specifics

7.1 General Public License

Like the original XFOIL, this project has been developed and released in accordance with the principles of the GPL. Among other things, one important point about GPL is that:

```
"This program is distributed in the hope that it will be useful, but
WITHOUT ANY WARRANTY; without even the implied warranty of
MERCHANTABILITY or FITNESS FOR A PARTICULAR PURPOSE. See the GNU
General Public License for more details."
```

7.2 XFOIL, AVL and XFLR5

XFLR5 has been developed based on XFOIL V6.94. Later additions to XFOIL have not been included in XFLR5.

Since the algorithms have been re-written and integrated in XFLR5, XFOIL does not need to be present on the computer for XFLR5 to run. No special links need to be declared.

XFLR5 does not use any of the AVL source code. The VLM algorithms have been developed and implemented independently.

For AVL files generated by XFLR5, the foil's file names will need to be checked, and it will also be necessary to check that the foil files are present in the directory together with the other AVL files.

7.3 Files and Registry

Running XFLR5 will generate two files in the installation directory:

- "XFLR5.set" which records user settings
- "XFLR5.log" which records the output of the foil and wing analysis

XFLR5 itself does not write anything in the registry, but the installation program will create shortcuts for the ".plr" and ".wpa" files. Users can choose to associate the foil ".dat" files to XFLR5, but since this extension is used by Windows for many different purposes, it has been deemed preferable to leave this choice to the user.

Registry shortcuts will be removed by the uninstall process.

7.4 Shortcuts

In an attempt to increase the user friendliness of the interface, shortcuts have been provided for most major commands, and are mentioned in the menus

Typing a first carriage return (↵) in a dialog box will select the OK or the default button, typing a second carriage return will activate this button.

Typing a first carriage return (↵) in the main window will select the 'Analyze' button, typing a second carriage return will activate this button.

7.5 Mouse input

All graphs, foils, and wings may be dragged and zoomed with the mouse. Using Ctrl+Left button in 3D view will cause rotation of the model.

These options however may not work correctly (or not at all) if the buttons are not set to the "Default" in the Windows Mouse interface.

For those computers without a mouse wheel nor a middle button, zooming can be achieved in all views by pressing the 'Z' key and moving the mouse.

7.6 Memory

One of the characteristics of both the foil and the wing analysis is to use a significant computer memory.

Operating points specifically store a large amount of data and lead to voluminous project files which will slow down Save & Load processes. It is however unnecessary to keep them, since the important data is also stored in the polar objects which do not require large memory resources.

7.7 Printing

Although XFLR5, as it is, offers some printing options, the implementation of more advanced capabilities would require significant work, and has not been, nor is expected to become, the primary concern of the on-going development.

For neat hardcopies, it is preferable and recommended to export the operating points or polars to a text file and process them in a spreadsheet.

8 Credits

Many thanks to Matthieu for his scientific advice and help, to Jean-Marc for his patient and comprehensive testing of the preliminary versions, and to all the others who have contributed by their input to improve XFLR5, especially Marc, Giorgio and Jean-Luc.

André Deperrois
June 2006

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- [1] James C. Sivells and Robert H. Neely, "Method for calculating wing characteristics by lifting line theory using nonlinear section lift data", April 1947, NACA Technical Note 1269.
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