XFLR5
Airfoil, Wing and Plane Analysis

John Sullivan

Description of XFLR5

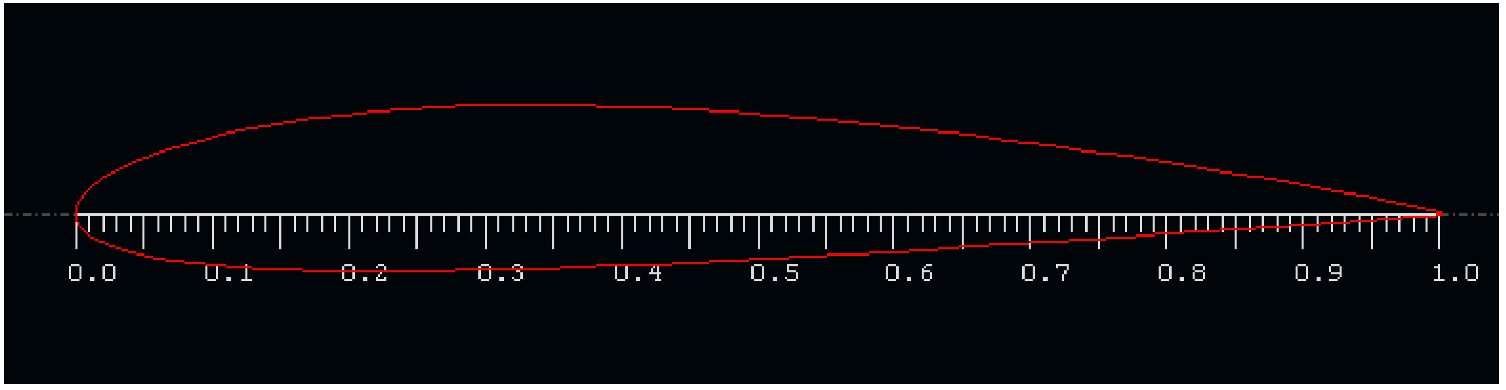
- XFLR5 is an analysis tool for airfoils, wings and planes. It includes:
  - XFoil's Direct and Inverse analysis capabilities
  - Wing design and analysis capabilities based on the Lifiting Line Theory, on the Vortex Lattice Method, and on a 3D Panel Method
- The airfoil analysis portion is based on the program XFOIL developed by Professor Mark Drela from MIT. XFLR5 has an updated GUI, so the operation of it is somewhat different than that of XFOIL.
- The XFLR5 program and Guidelines can be downloaded from the project web site:
- http://sourceforge.net/projects/xflr5/
- http://www.xflr5.com/xflr5.htm (links to related material)
- More information on XFOIL is available at http://web.mit.edu/drela/Public/web/xfoil/ and a simple tutorial can be found on the course website in the file XFOIL_tutorial.pdf.

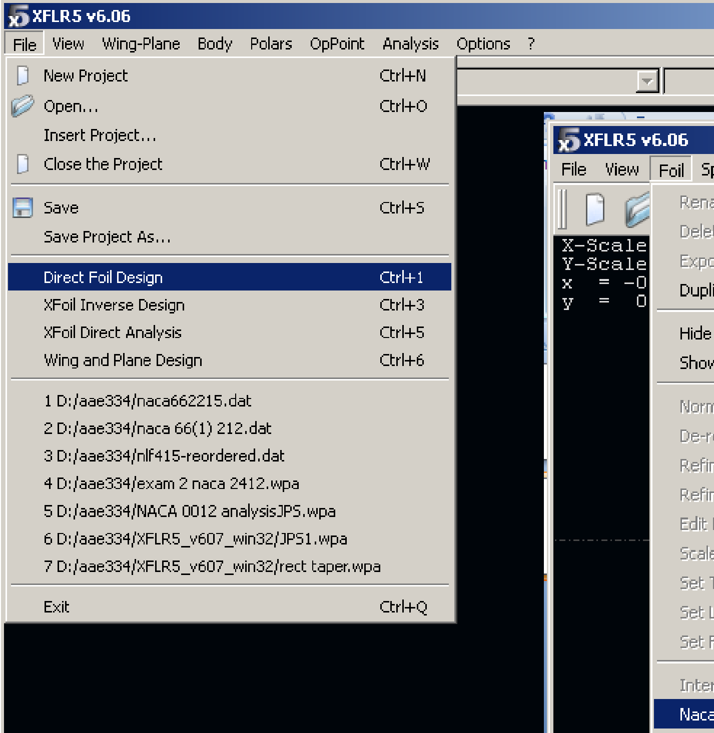
XFLR5 Examples

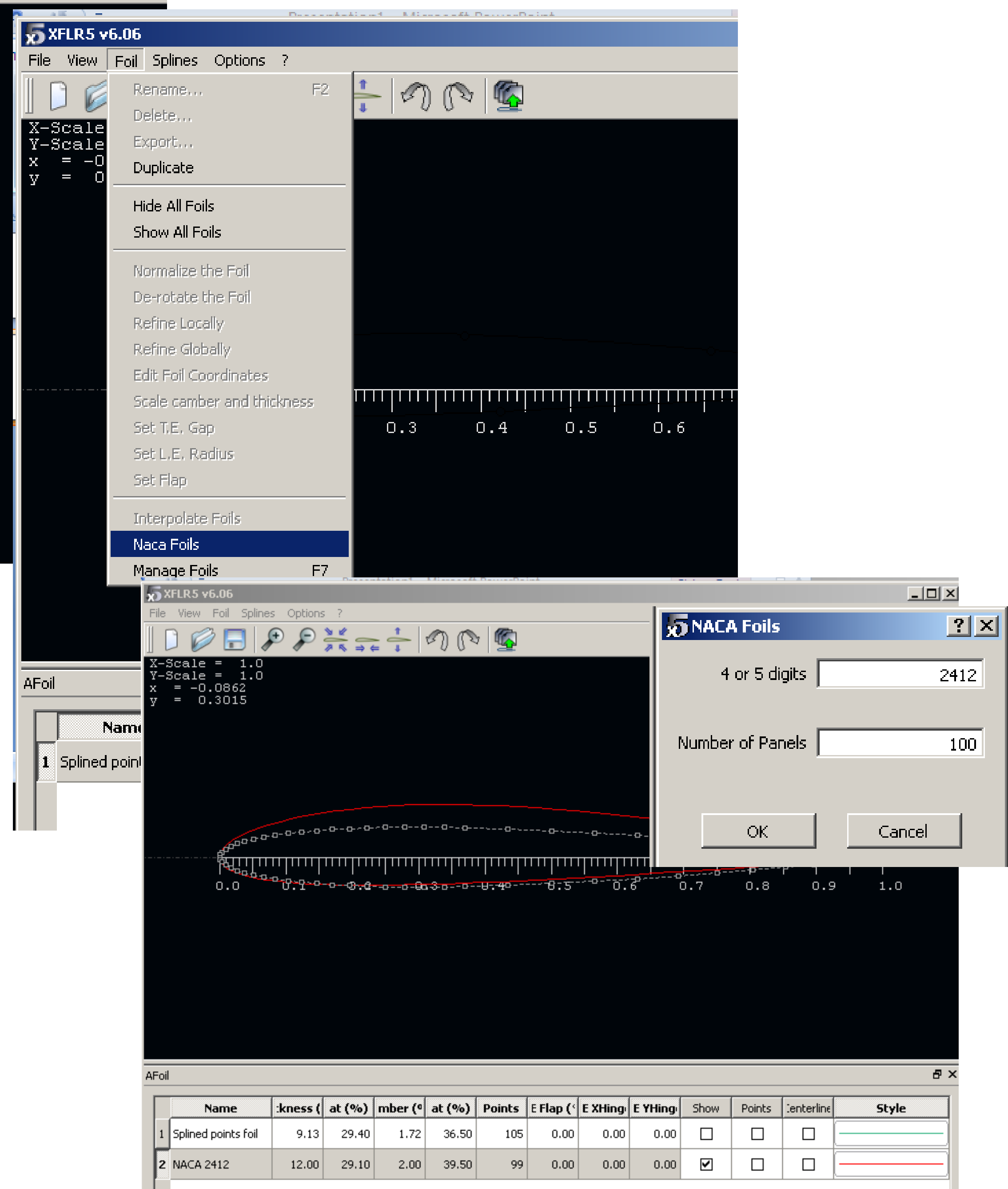
1. NACA 2415
2. NACA 66(2)215 laminar flow airfoil
3. Rectangular Wing
   a) Lifting Line Theory
   b) 3D Panel method

Example #1 NACA 4 digit Airfoil Analysis

- NACA 2412
- Reynolds number 1 million to 10 million in steps of 1 million
- Angle of attack -5 to 10 degrees in steps of .2 degrees

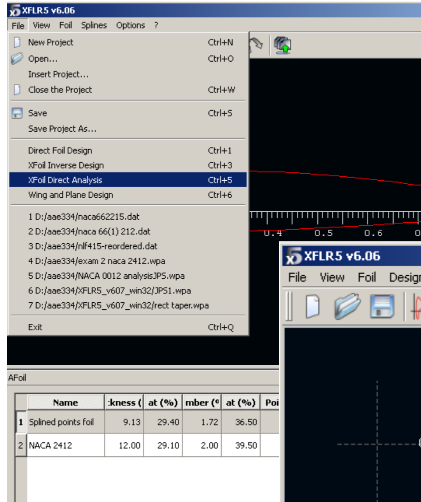




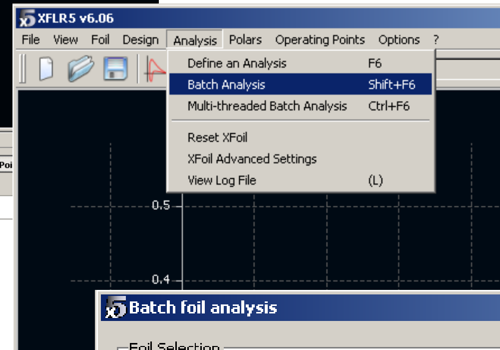


- Start XFLR5
- Click <File><Direct Foil Design>
- Click <Foil Design><Naca Foils>
- Enter the 4- or 5-digit name of the airfoil and the number of panels to use

•Click <File> <XFOIL <Direct Analysis>



•Click <Analysis> <BatchAnalysis>.



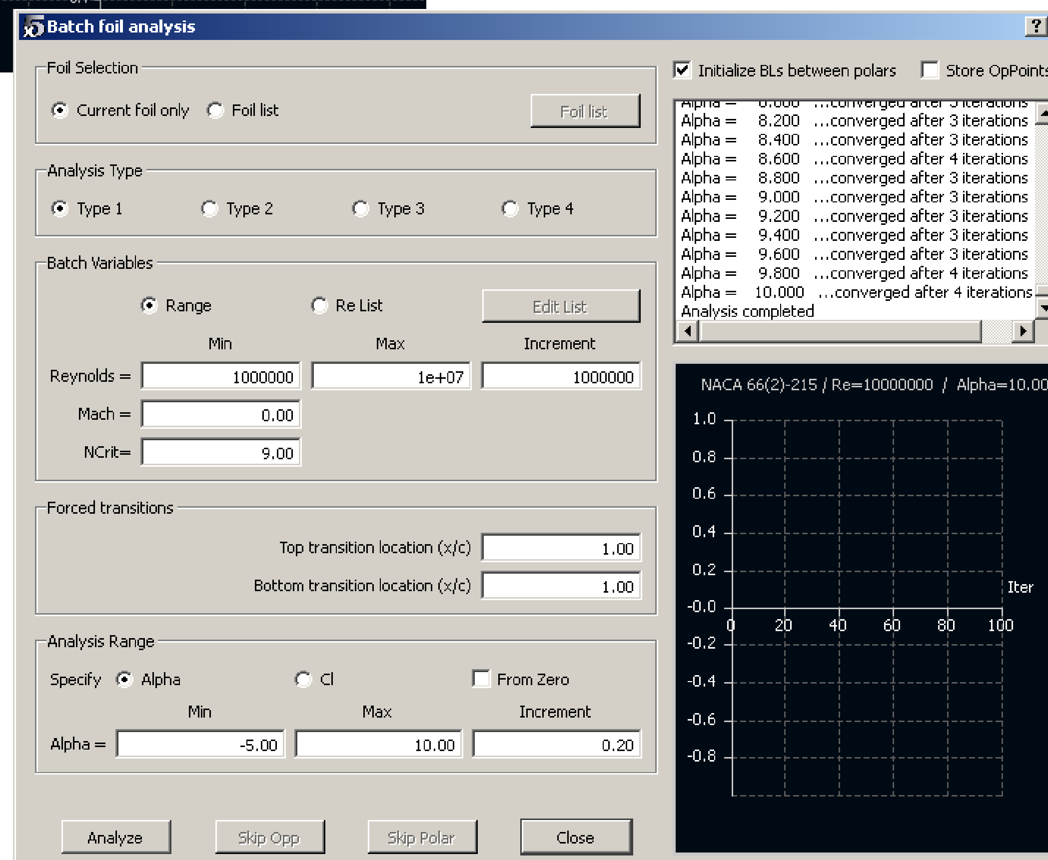
•Choose Type 1.

•Enter Reynolds number, Mach number and transition information.

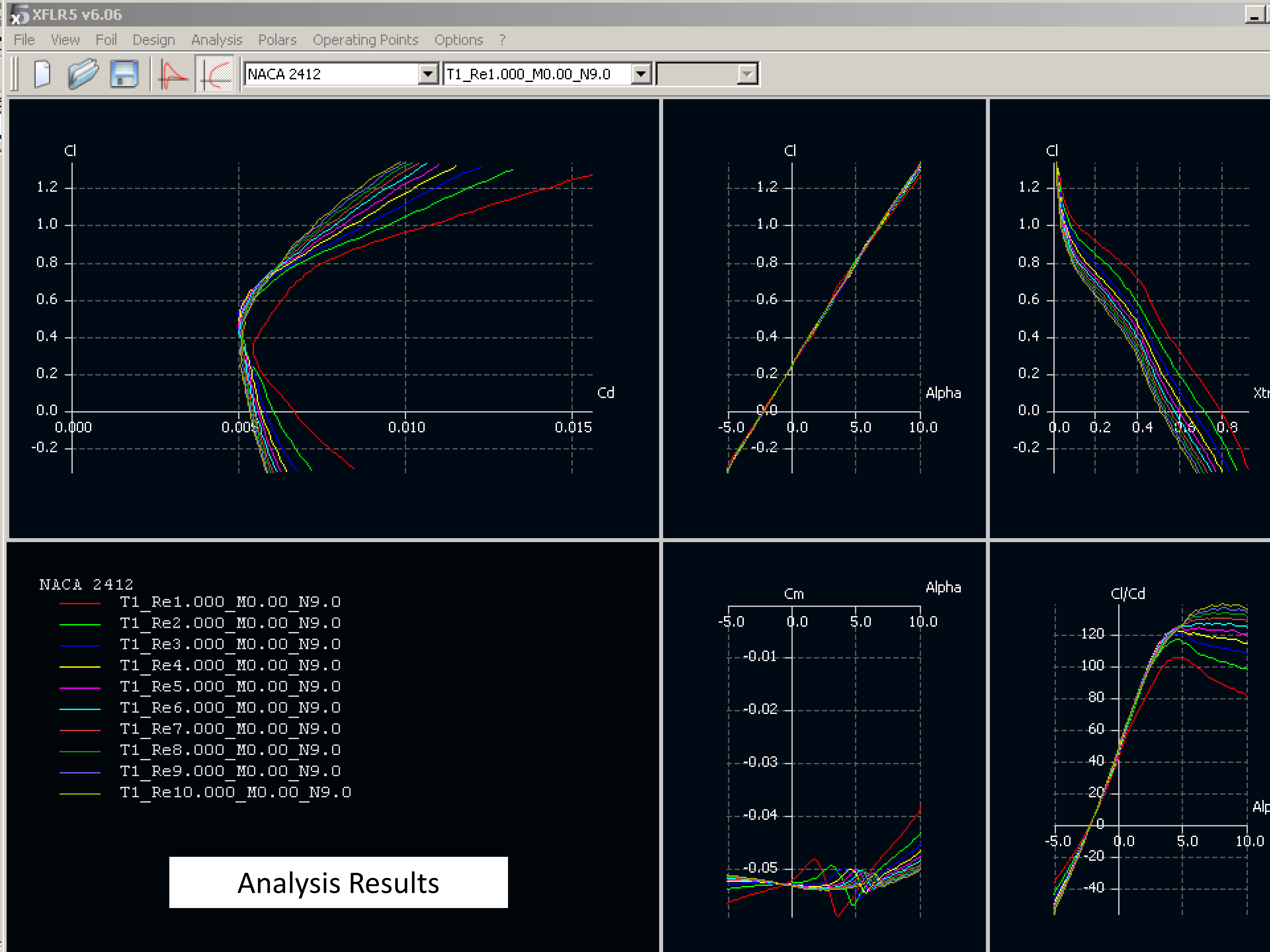
•Enter angles of attack

•

Click the <Analyze> button.



The program runs through an iterative procedure to solve the problem at each angle of attack Click close when finished


XFLR5 v6.06
File View Foil Design Analysis Polars Operating Points Options ?
NACA 2412
T1_Re1.000_M0.00_N9.0
Cl
Cd
0.000 0.005 0.010 0.015
1.2 1.0 0.8 0.6 0.4 0.2 0.0 -0.2
Alpha
-5.0 0.0 5.0 10.0
Xtr
0.0 0.2 0.4 0.6 0.8
Cm
-0.01 -0.02 -0.03 -0.04 -0.05
Cl/Cd
120 100 80 60 40 20 0 -20 -40
NACA 2412
T1_Re1.000_M0.00_N9.0
T1_Re2.000_M0.00_N9.0
T1_Re3.000_M0.00_N9.0
T1_Re4.000_M0.00_N9.0
T1_Re5.000_M0.00_N9.0
T1_Re6.000_M0.00_N9.0
T1_Re7.000_M0.00_N9.0
T1_Re8.000_M0.00_N9.0
T1_Re9.000_M0.00_N9.0
T1_Re10.000_M0.00_N9.0
Analysis Results

Pressure Distribution

1. Click the pressure distribution icon

3. Change Angle of Attack

2. Click analyze

- The program may not converge for a given angle of attack if the solution is particularly complex or if the change from the initial guess or the previous solution is too large. You can increase the maximum number of iterations in the <Analysis><XFOIL Advanced Settings> menu. If there are a small number of angles of attack for which the solution did not converge, that is OK. Just realize that the results for those angles of attack are more unreliable.

- The airfoil is shown in the bottom part of the window at the angle of attack at the end of the sequence you chose. In the upper part of the window is shown the pressure coefficient distribution. To see Cp using an inviscid analysis (panel method), choose <Operating Points><Cp Graph><Show Inviscid Curve> (or <right click> on the Cp graph instead of <Operating Points>). The inviscid Cp distribution shows up as a dashed line, while the solid line shows Cp accounting for viscous boundary layer effects. You can choose a particular angle of attack by clicking on the button on the far right side of the tool bar.

- Check the box in the XDirect pane for Show Pressure to see the local pressure distribution on the airfoil shown as force arrows. Check the Show Boundary Layer box to see the boundary layer thickness on the airfoil surface. Check the Animate box to see a sweep through the angles of attack and to watch the results change.

- Click <Polars><Polar Graphs><All Polar Graphs> to see the five polar plots. The menu that the choice <All Polar Graphs> is on shows what is plotted as figures (1) through (5). (Note: There is a short cut button in the tool bar at the top to switch between polars and the Cp plot.)

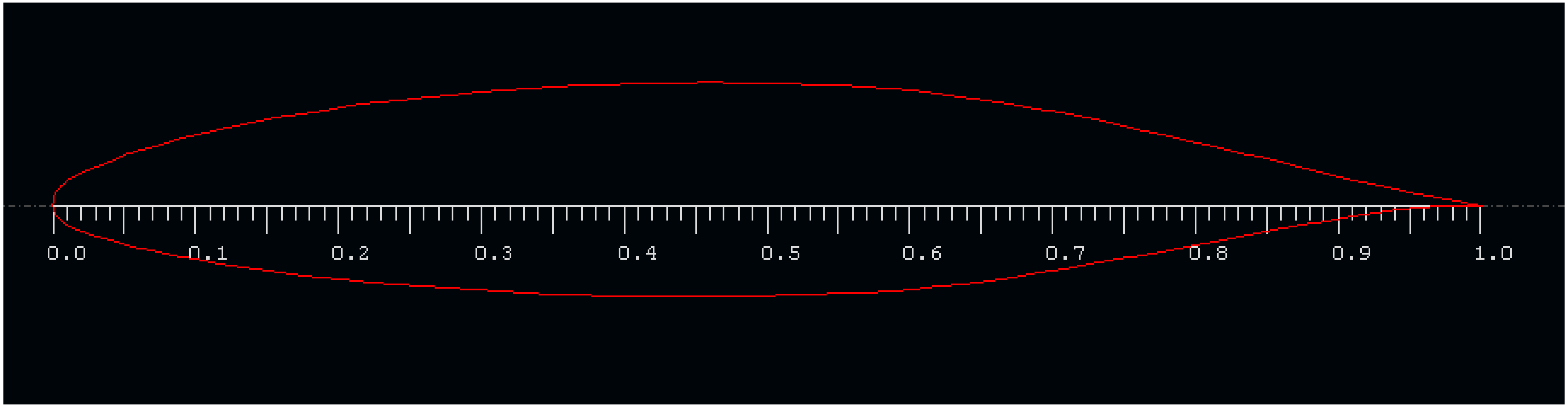
- You can use the mouse to zoom in and out and to translate any of the plots.

- To save a plot choose <Right Click><Save View to Image File>. Possible file formats are bitmap, jpeg and png.

- To plot other variables computed by XFOIL, on the Cp plot choose <Right Click><Cp Graph><Current XFOIL Results> and then the name of the variable you want to plot, e.g. <Skin Friction Coefficient>. (The variables D* and Theta refer to $d^* = d_1$ = boundary layer displacement thickness and $q = d_2$ = boundary layer momentum thickness.)

Example #2 NACA 66(2)-215 Airfoil Analysis

- NACA 66(2)-215 laminar airfoil
- Reynolds number 1 million to 10 million in steps of 1 million
- Angle of attack -5 to 10 degrees in steps of .2 degrees



Airfoil Coordinates for XFLR5

- XFLR5 reads coordinates from a *.dat file
- The points must be in (x,y) pairs, starting at the trailing edge (TE), going to the leading edge (LE), and back to the TE. The points may go over the upper surface and back along the lower surface, or vice versa (the code can figure that out).
- The first line is the airfoil name
- A good source of airfoil data is:
- http://www.ae.illinois.edu/m-selig/ads/coord_database.html
- Note that some of this data is in the wrong format and must be reordered

Example file naca662215.dat

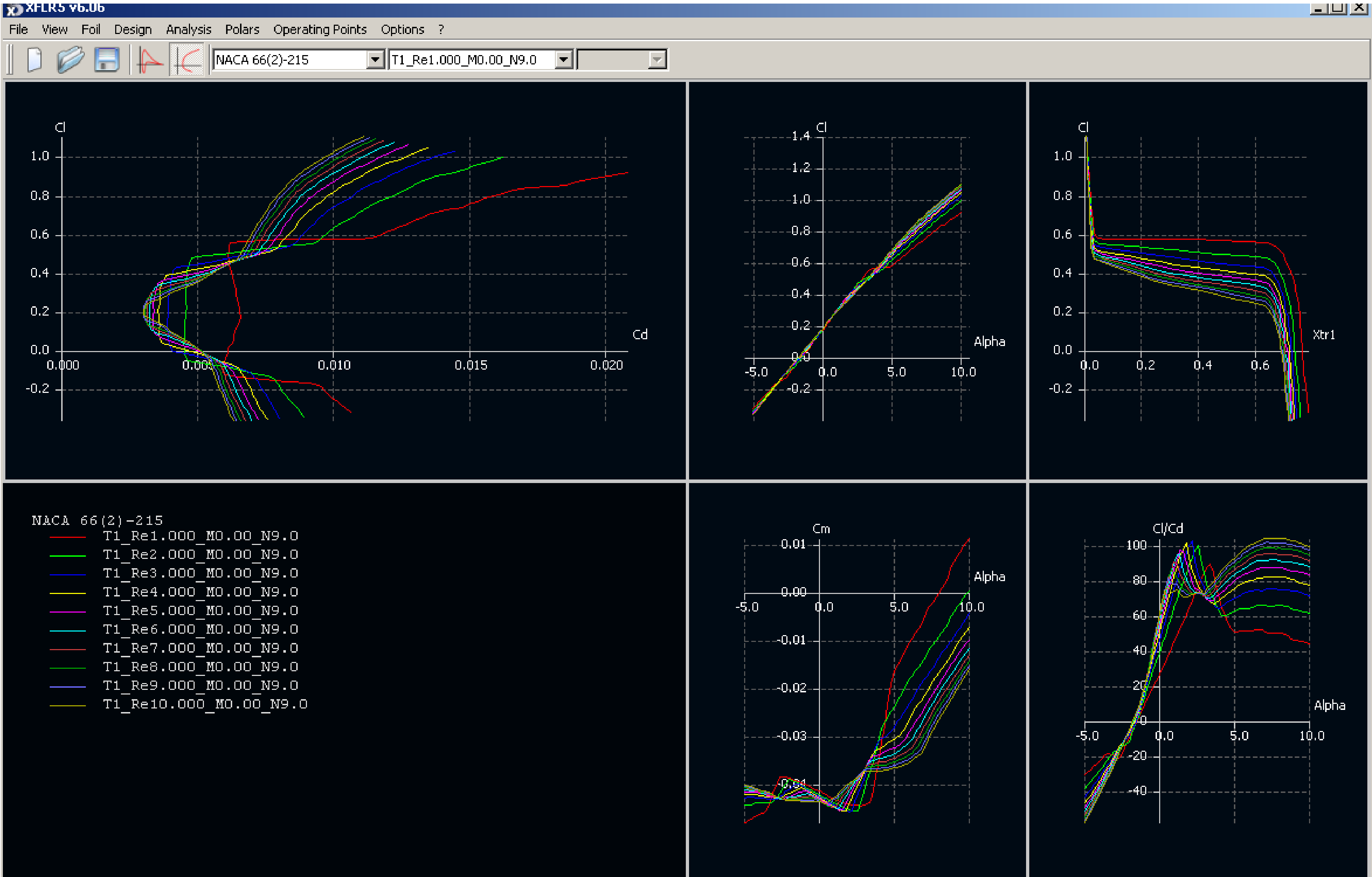
```
NACA 66(2)-215
    1.000000   0.000000
    0.993359   0.001014
    0.982368   0.002802
    0.969897   0.004996
    0.955711   0.007707
    0.939801   0.011019
    0.922598   0.014866
    0.904739   0.019047
    0.886614   0.023373
    0.868296   0.027778
    0.849849   0.032225
…………………………………
…………………………………
    0.909485  -0.008053
    0.926545  -0.005413
    0.942539  -0.003293
    0.957211  -0.001775
    0.970490  -0.000802
    0.982471  -0.000252
    0.993316  -0.000019
    1.000000   0.000000
```

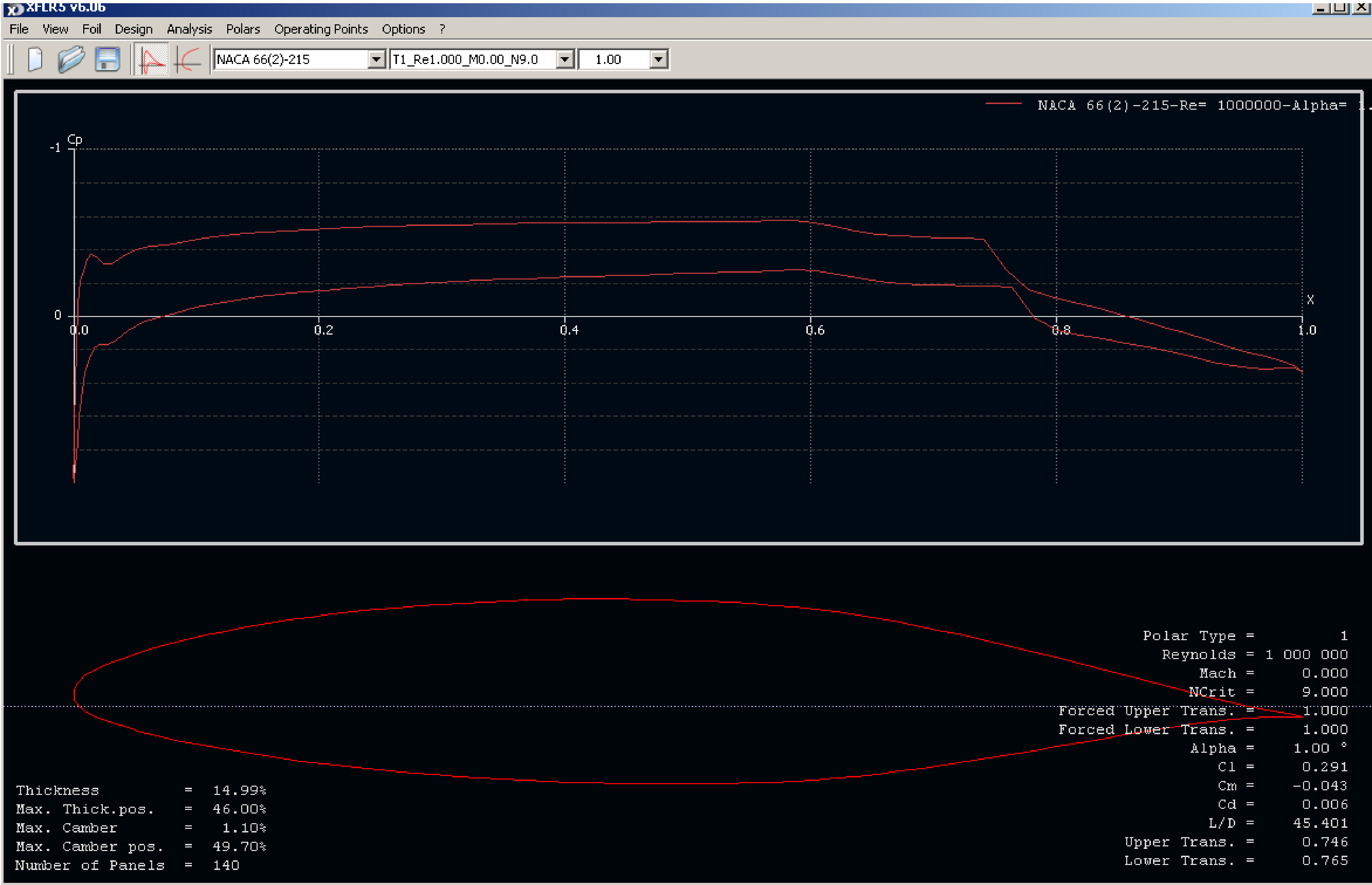
Airfoil Analysis

- Click <File> < new project>
- Click <File> < open> naca662215.dat
- Click <File> < direct foil design> to see airfoil
- Click <File> < Xfoil direct analysis>
- Click <Analysis> <BatchAnalysis>.
- Choose Type 1.
- Enter Reynolds number, Mach number and transition information.
- Enter angles of attack
-
- Click the <Analyze> button.

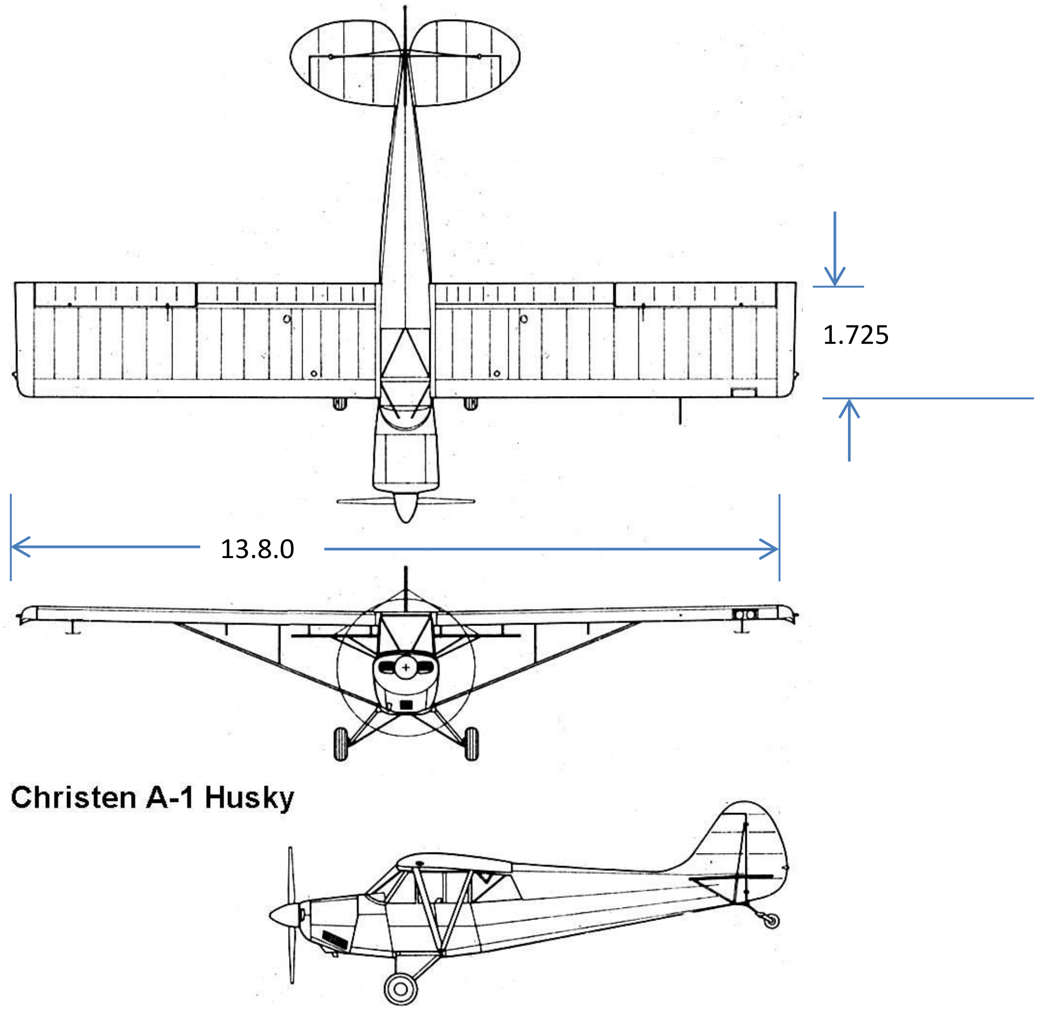
Same as Example #1

NACA 66(2)-215


XFLR5 v6.06
File View Foil Design Analysis Polars Operating Points Options ?
NACA 66(2)-215
T1_Re1.000_M0.00_N9.0
Cl
Cd
Alpha
Xtr1
Cm
Cl/Cd
NACA 66(2)-215
T1_Re1.000_M0.00_N9.0
T1_Re2.000_M0.00_N9.0
T1_Re3.000_M0.00_N9.0
T1_Re4.000_M0.00_N9.0
T1_Re5.000_M0.00_N9.0
T1_Re6.000_M0.00_N9.0
T1_Re7.000_M0.00_N9.0
T1_Re8.000_M0.00_N9.0
T1_Re9.000_M0.00_N9.0
T1_Re10.000_M0.00_N9.0


XFLR5 v6.06
File
View
Foil
Design
Analysis
Polars
Operating Points
Options
?
NACA 66(2)-215
T1_Re1.000_M0.00_N9.0
1.00
NACA 66(2)-215-Re= 1000000-Alpha= 1.
Cp
-1
0
X
0.0
0.2
0.4
0.6
0.8
1.0
Polar Type = 1
Reynolds = 1 000 000
Mach = 0.000
NCrit = 9.000
Forced Upper Trans. = 1.000
Forced Lower Trans. = 1.000
Alpha = 1.00 °
Cl = 0.291
Cm = -0.043
Cd = 0.006
L/D = 45.401
Upper Trans. = 0.746
Lower Trans. = 0.765
Thickness = 14.99%
Max. Thick.pos. = 46.00%
Max. Camber = 1.10%
Max. Camber pos. = 49.70%
Number of Panels = 140

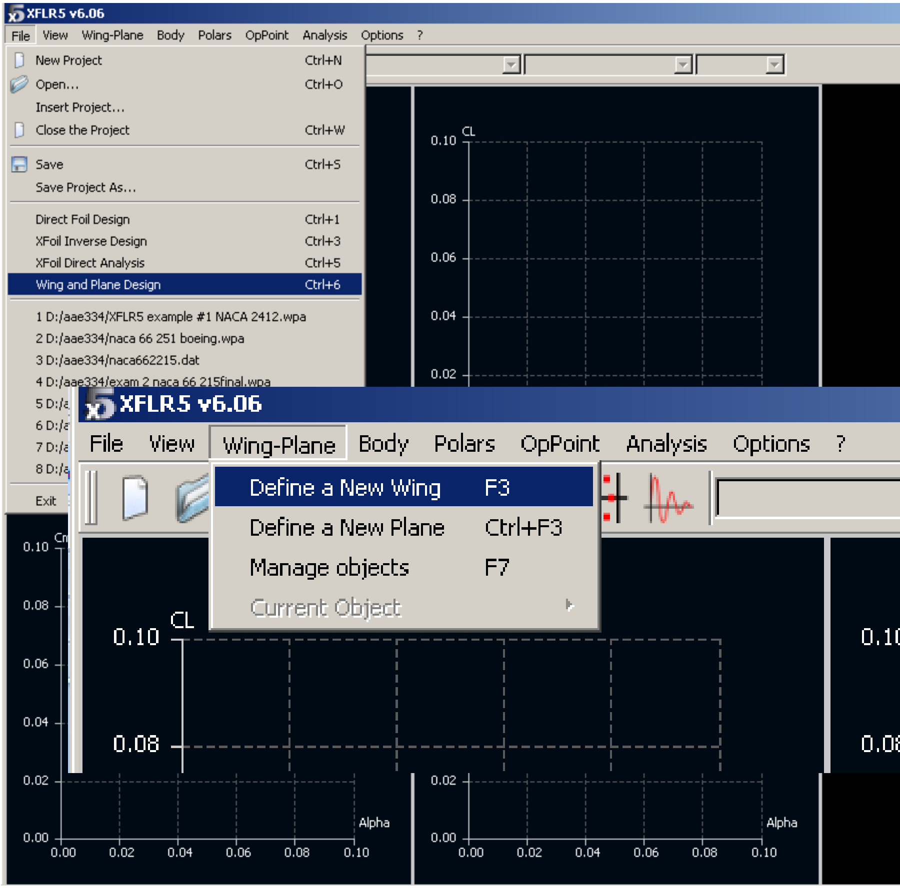
Example #3a NACA 2415 Rectangular Wing Lifting Line Theory

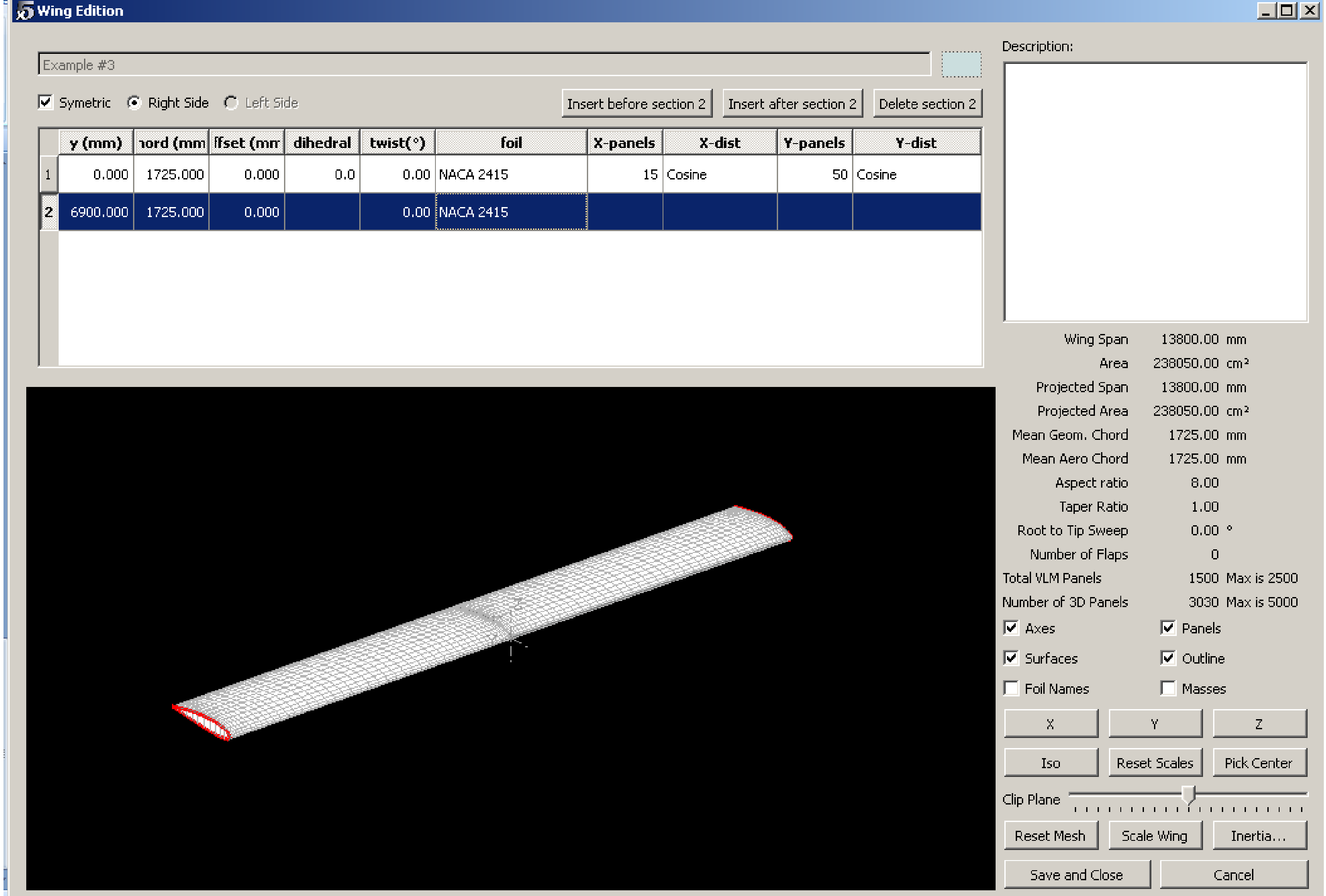


Wing Analysis

- Click <File> < new project>
- Run Example #1 NACA 2415 airfoil
- Click <file> <wing and plane Design
- Click <wing and plane<new wing design>

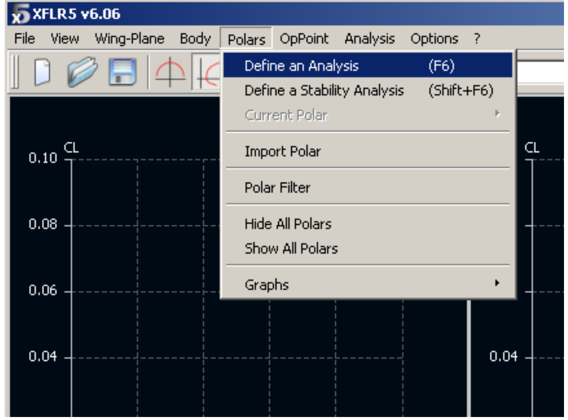
- The wing edition window pops up

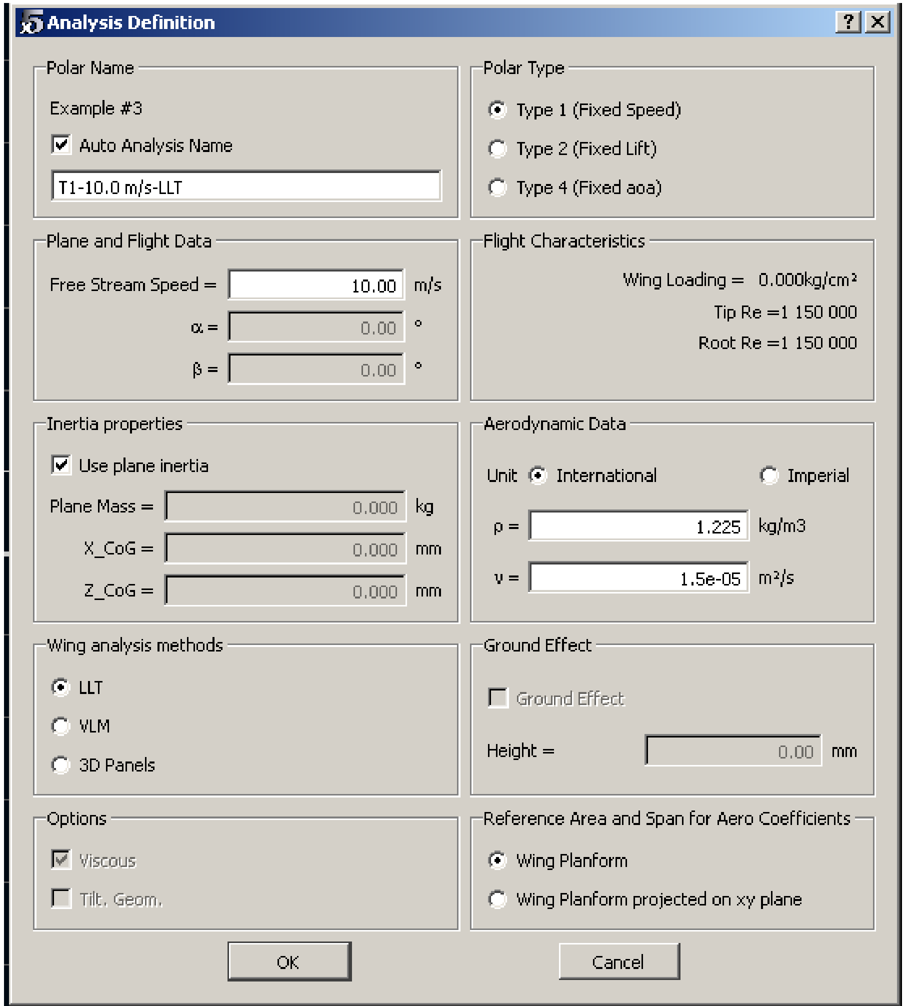




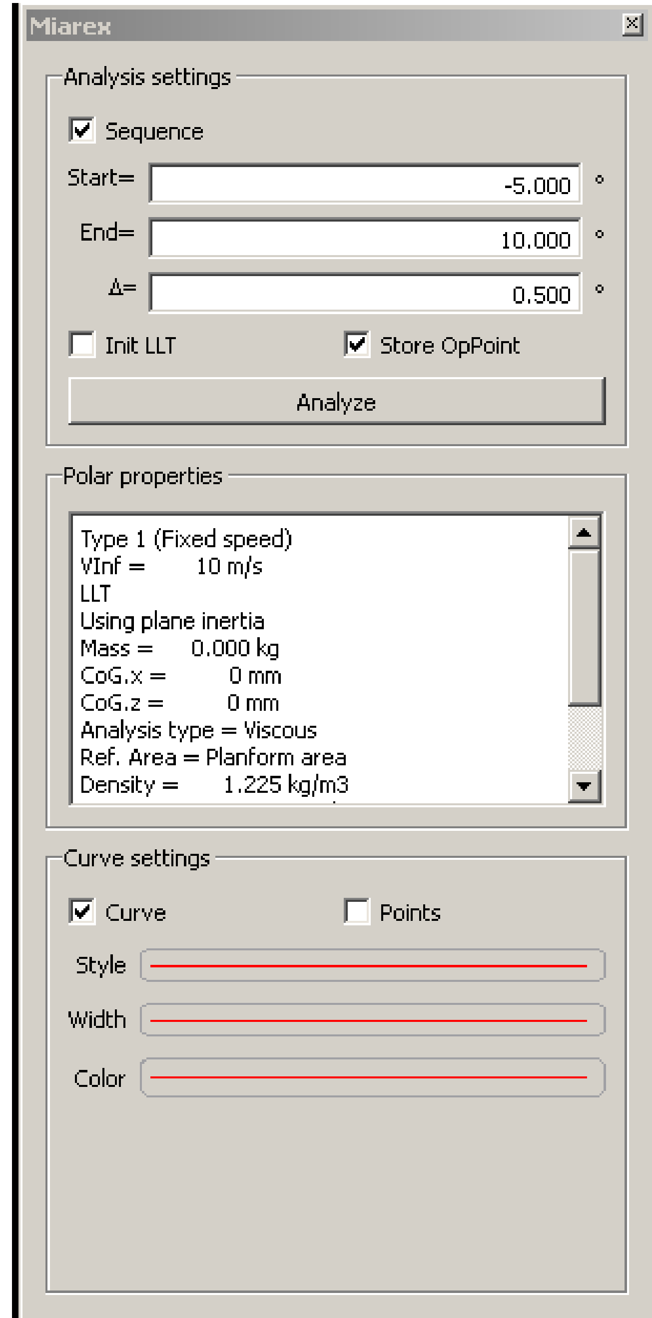
- Click Symmetry and right side
- Add dimensions for the right half wing
- Click <foil> and choose NACA2415
- Add the panel numbers and click the distribution and choose cosine
- Check calculated quantities on right side
- Click ,save and close.

- Click <polars><define analysis
- In pop-up window
  - Choose auto name,Type 1
  - Input Free stream speed
    - Check calculated Re numbers to be sure they are in the range of the airfoil analysis . You will get an error if they are out of range.
  - Choose international units
  - Choose LLT – lifting Line Theory
- Clikck OK

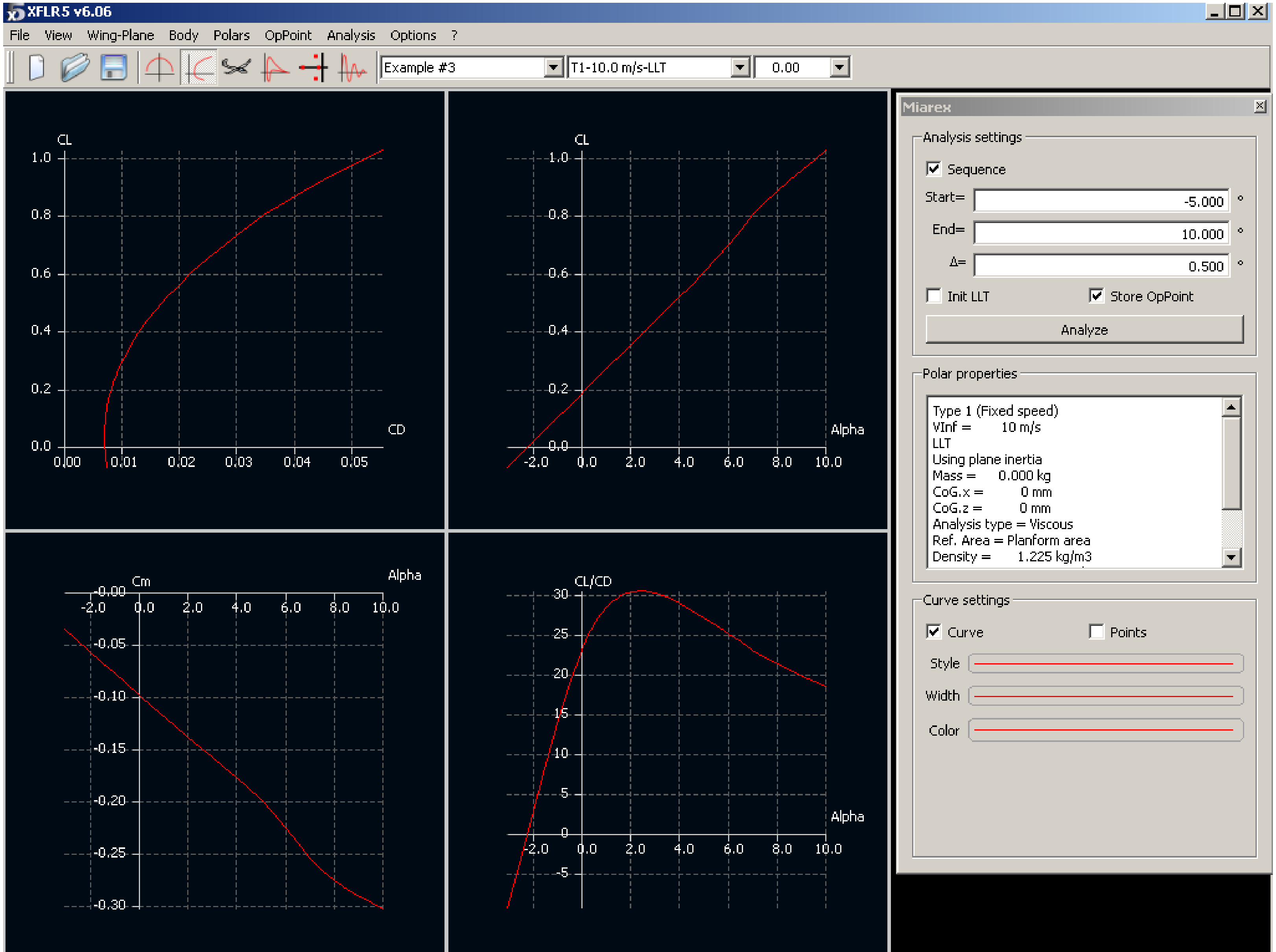


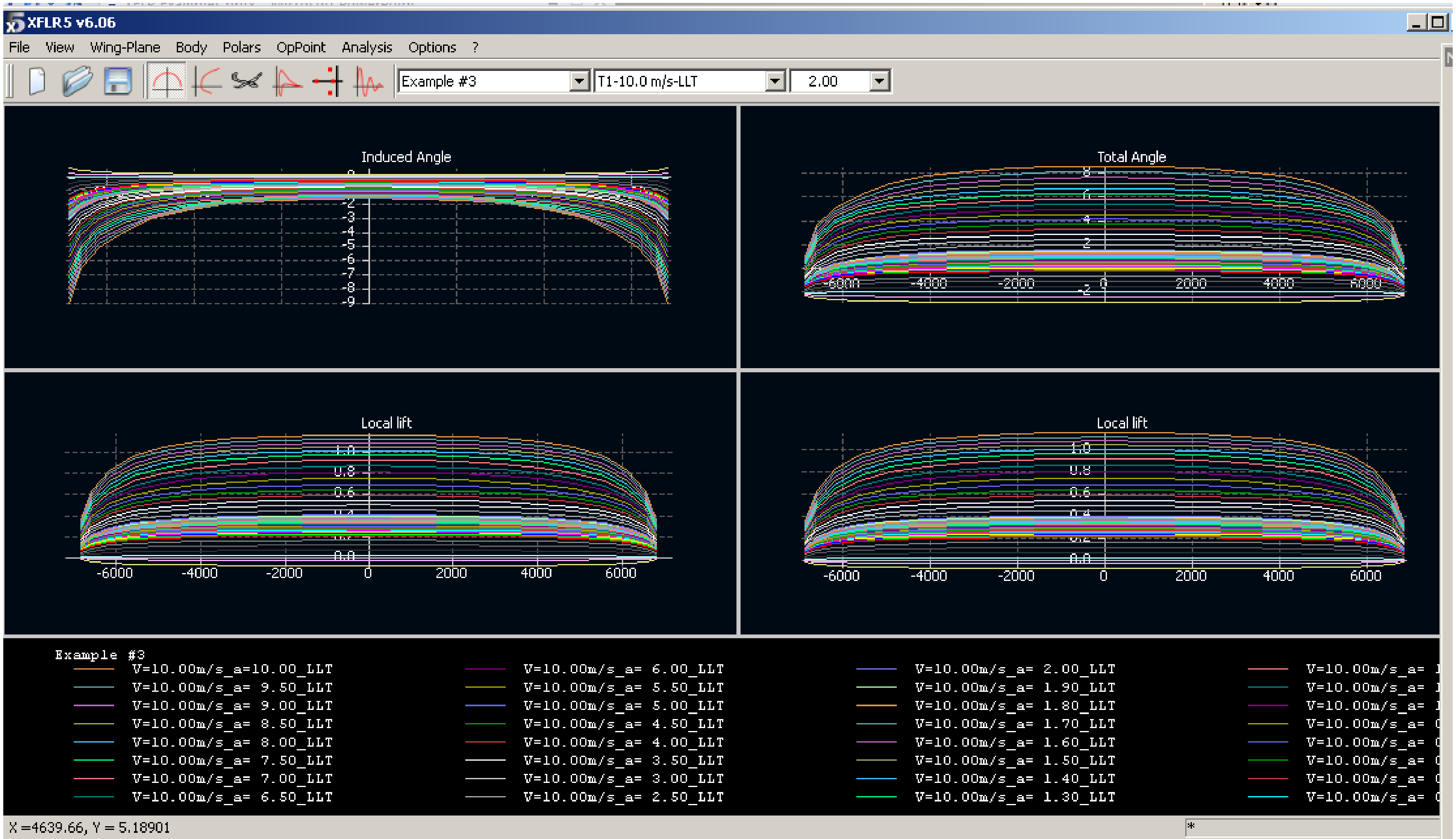


- On Pop-up
- Set angle of attack range
- Click Analyze

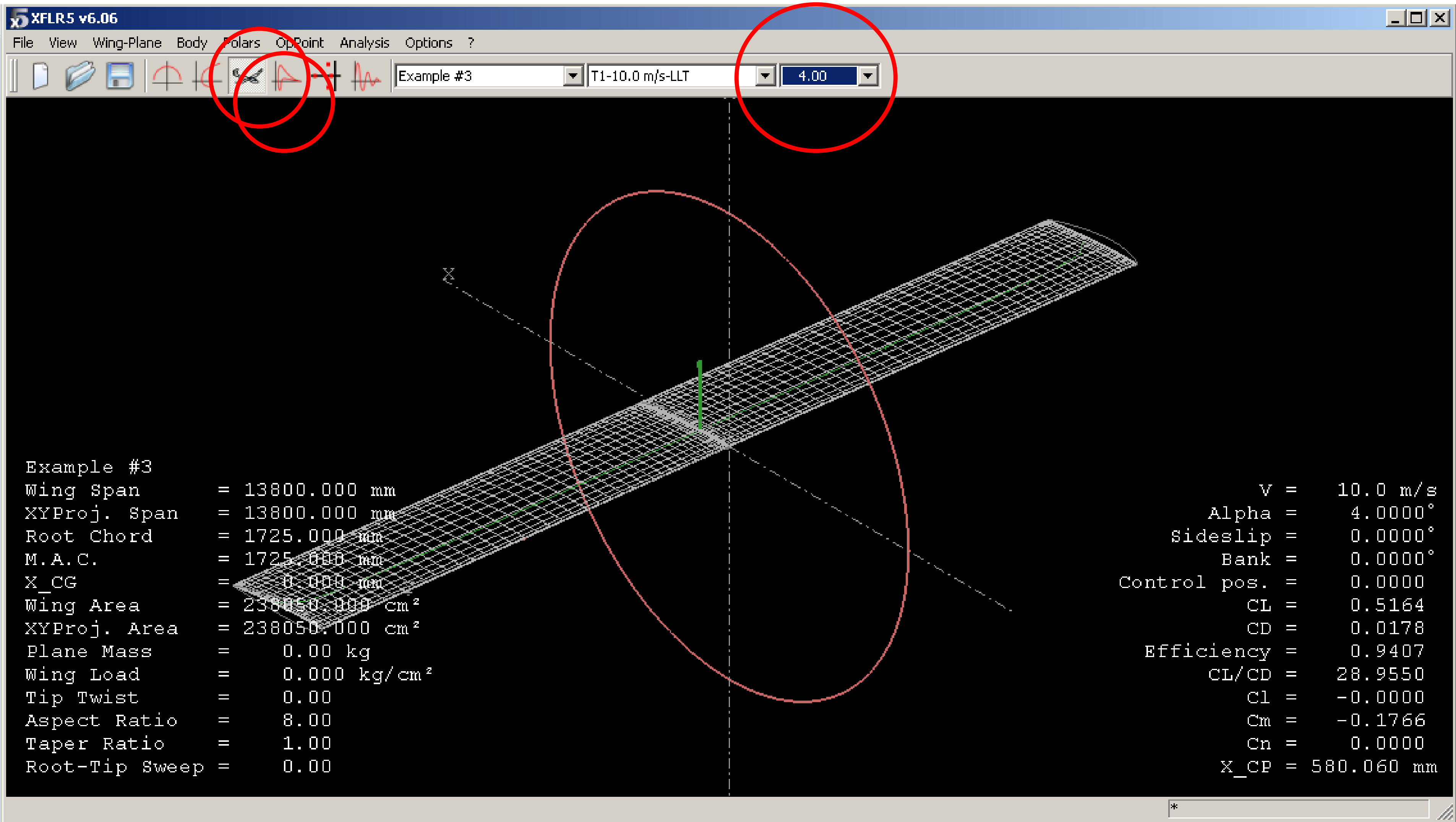


Polars


XFLR5 v6.06
File View Wing-Plane Body Polars OpPoint Analysis Options ?
Example #3
T1-10.0 m/s-LLT
0.00
CL
CD
Alpha
Cm
CL/CD
Miarex
Analysis settings
Sequence
Start= -5.000 °
End= 10.000 °
Δ= 0.500 °
Init LLT
Store OpPoint
Analyze
Polar properties
Type 1 (Fixed speed)
VInf = 10 m/s
LLT
Using plane inertia
Mass = 0.000 kg
CoG.x = 0 mm
CoG.z = 0 mm
Analysis type = Viscous
Ref. Area = Planform area
Density = 1.225 kg/m3
Curve settings
Curve
Points
Style
Width
Color


XFLR5 v6.06
File View Wing-Plane Body Polars OpPoint Analysis Options ?
Example #3
T1-10.0 m/s-LLT
2.00
Induced Angle
Total Angle
Local lift
Local lift
Example #3
V=10.00m/s_a=10.00_LLT
V=10.00m/s_a= 9.50_LLT
V=10.00m/s_a= 9.00_LLT
V=10.00m/s_a= 8.50_LLT
V=10.00m/s_a= 8.00_LLT
V=10.00m/s_a= 7.50_LLT
V=10.00m/s_a= 7.00_LLT
V=10.00m/s_a= 6.50_LLT
V=10.00m/s_a= 6.00_LLT
V=10.00m/s_a= 5.50_LLT
V=10.00m/s_a= 5.00_LLT
V=10.00m/s_a= 4.50_LLT
V=10.00m/s_a= 4.00_LLT
V=10.00m/s_a= 3.50_LLT
V=10.00m/s_a= 3.00_LLT
V=10.00m/s_a= 2.50_LLT
V=10.00m/s_a= 2.00_LLT
V=10.00m/s_a= 1.90_LLT
V=10.00m/s_a= 1.80_LLT
V=10.00m/s_a= 1.70_LLT
V=10.00m/s_a= 1.60_LLT
V=10.00m/s_a= 1.50_LLT
V=10.00m/s_a= 1.40_LLT
V=10.00m/s_a= 1.30_LLT
X =4639.66, Y = 5.18901

Operating Point



Right Click on graph, choose current op point, export
Generates file of data

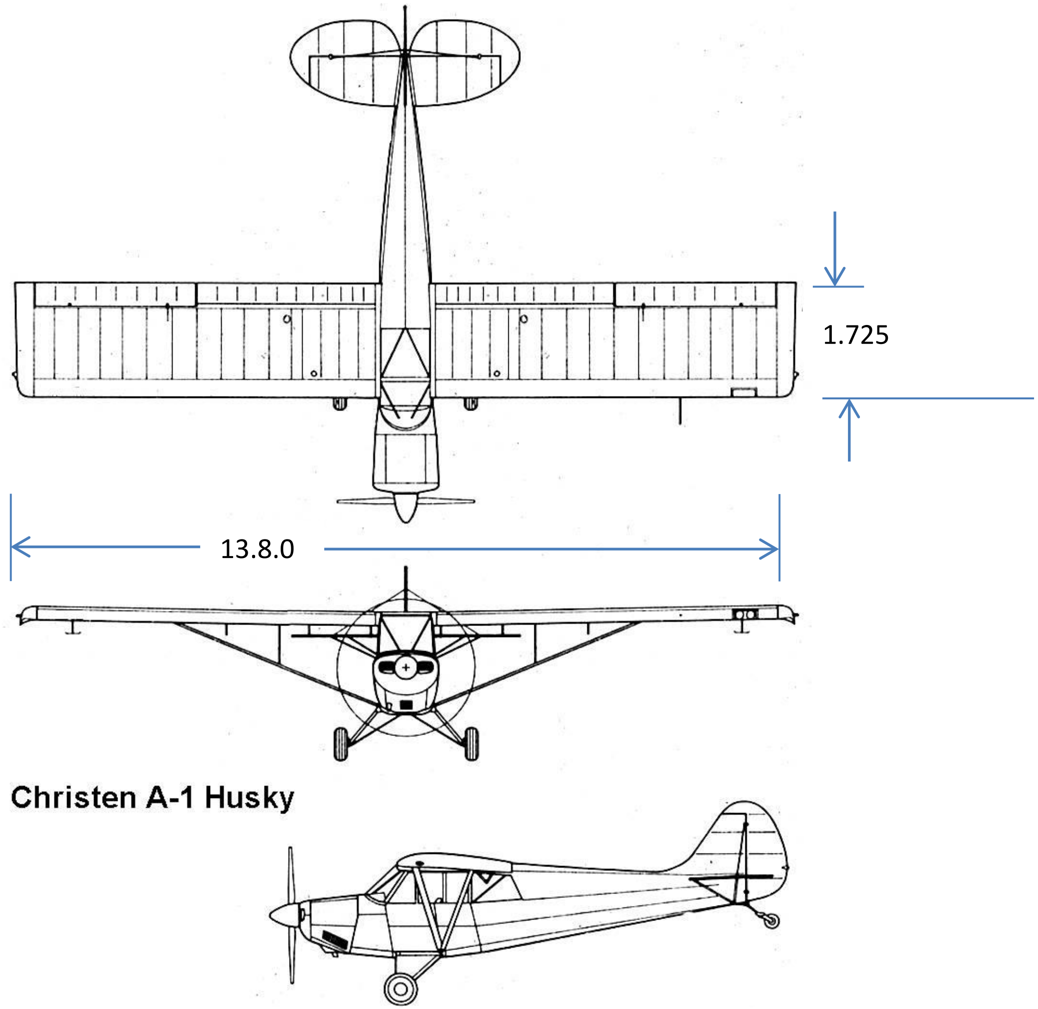
XFLR5 v6.06

Example #3
T1-10.0 m/s-LLT
QInf = 10.000000 m/s
Alpha = 4.000000
Beta = 0.000°
Phi = 0.000°
Ctrl = 0.000
CL = 0.516417
Cy = 0.000000
Cd = 0.017835 ICd = 0.011280 PCd = 0.006555
Cl = -3.62042e-17
Cm = -0.176573
ICn = 0.000000 PCn = 0.000000
XCP = 0.580060 YCP = 0.000000
XNP = 0.000000
Bend. = 1182.365112

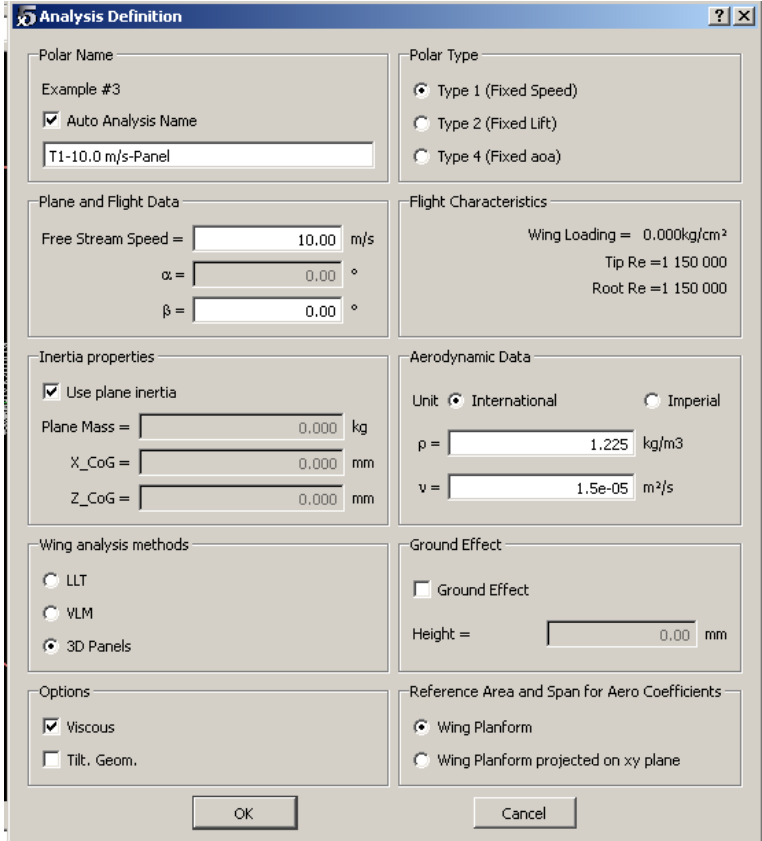
Example #3

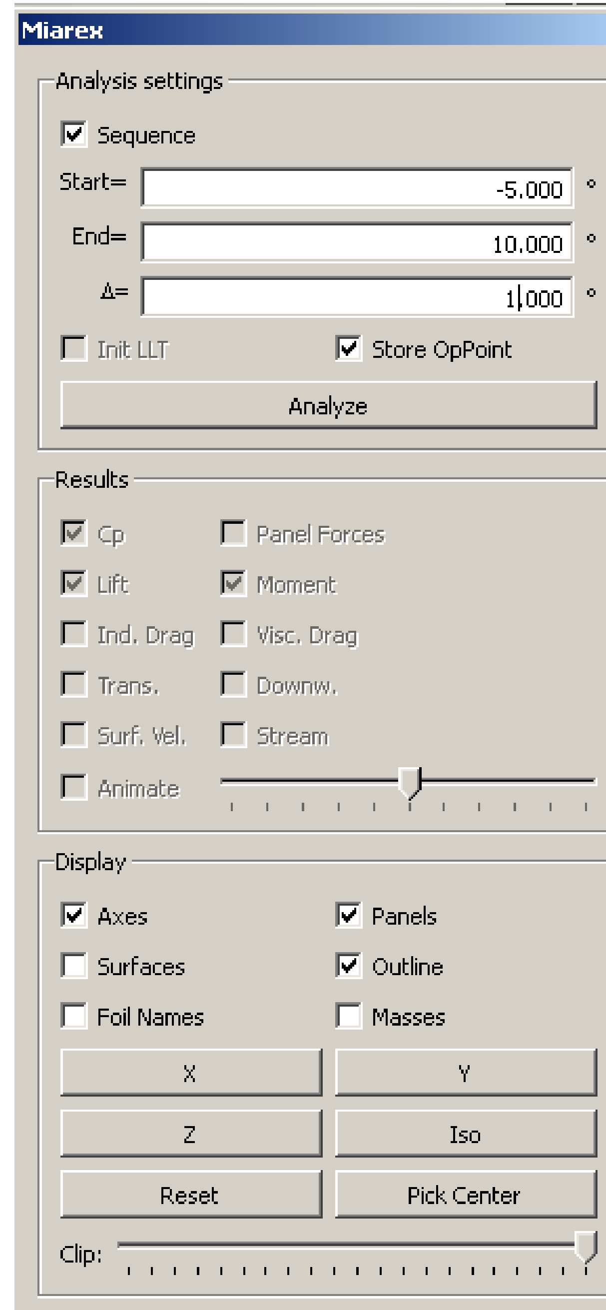
y-span	Chord	Ai	Cl	PCd	ICd	CmGeom	CmAirf	XTrtop	XTrBot	XCP	BM
-6.8150	1.7250	-4.594	0.172674	0.006612	0.013845	-0.094896	-0.051747	0.6109	0.4782	0.5500	0.0000
-6.5623	1.7250	-3.350	0.308041	0.006376	0.018009	-0.127697	-0.050674	0.5468	0.6522	0.4113	0.7787
-6.1479	1.7250	-2.458	0.404226	0.006369	0.017345	-0.150668	-0.049605	0.5049	0.7636	0.3682	6.5532
-5.5822	1.7250	-1.848	0.469725	0.006398	0.015152	-0.166086	-0.048677	0.4806	0.8337	0.3484	26.2776
-4.8790	1.7250	-1.438	0.513307	0.006472	0.012887	-0.176195	-0.047924	0.4645	0.8757	0.3377	72.9358
-4.0557	1.7250	-1.167	0.541930	0.006549	0.011034	-0.182737	-0.047339	0.4537	0.8999	0.3314	161.6461
-3.1325	1.7250	-0.989	0.560369	0.006636	0.009673	-0.186913	-0.046927	0.4451	0.9141	0.3276	307.2766
-2.1322	1.7250	-0.882	0.571906	0.006668	0.008799	-0.189599	-0.046744	0.4414	0.9222	0.3255	522.0347
-1.0794	1.7250	-0.823	0.578211	0.006685	0.008303	-0.191067	-0.046644	0.4394	0.9266	0.3244	813.3718
-0.0000	1.7250	-0.804	0.580211	0.006690	0.008143	-0.191532	-0.046612	0.4388	0.9280	0.3240	1182.3651
1.0794	1.7250	-0.823	0.578211	0.006685	0.008303	-0.191067	-0.046644	0.4394	0.9266	0.3244	813.3718
2.1322	1.7250	-0.882	0.571906	0.006668	0.008799	-0.189599	-0.046744	0.4414	0.9222	0.3255	522.0347
3.1325	1.7250	-0.989	0.560369	0.006636	0.009673	-0.186913	-0.046927	0.4451	0.9141	0.3276	307.2766
4.0557	1.7250	-1.167	0.541930	0.006549	0.011034	-0.182737	-0.047339	0.4537	0.8999	0.3314	161.6461
4.8790	1.7250	-1.438	0.513307	0.006472	0.012887	-0.176195	-0.047924	0.4645	0.8757	0.3377	72.9358
5.5822	1.7250	-1.848	0.469725	0.006398	0.015152	-0.166086	-0.048677	0.4806	0.8337	0.3484	26.2776
6.1479	1.7250	-2.458	0.404226	0.006369	0.017345	-0.150668	-0.049605	0.5049	0.7636	0.3682	6.5532
6.5623	1.7250	-3.350	0.308041	0.006376	0.018009	-0.127697	-0.050674	0.5468	0.6522	0.4113	0.7787
6.8150	1.7250	-4.594	0.172674	0.006612	0.013845	-0.094896	-0.051747	0.6109	0.4782	0.5500	0.0000

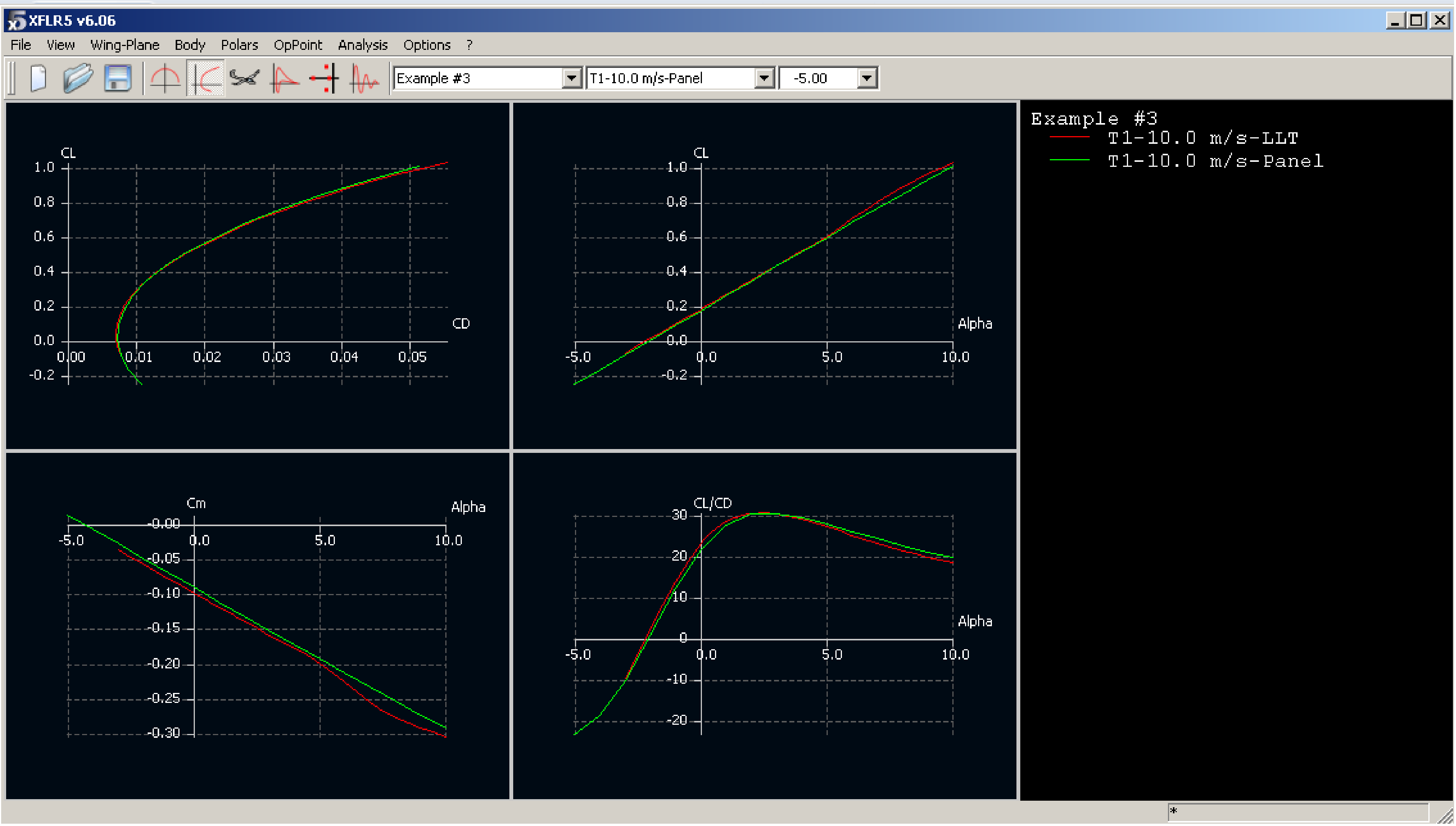
Example #3b NACA 2415 Rectangular Wing 3D Panel Method

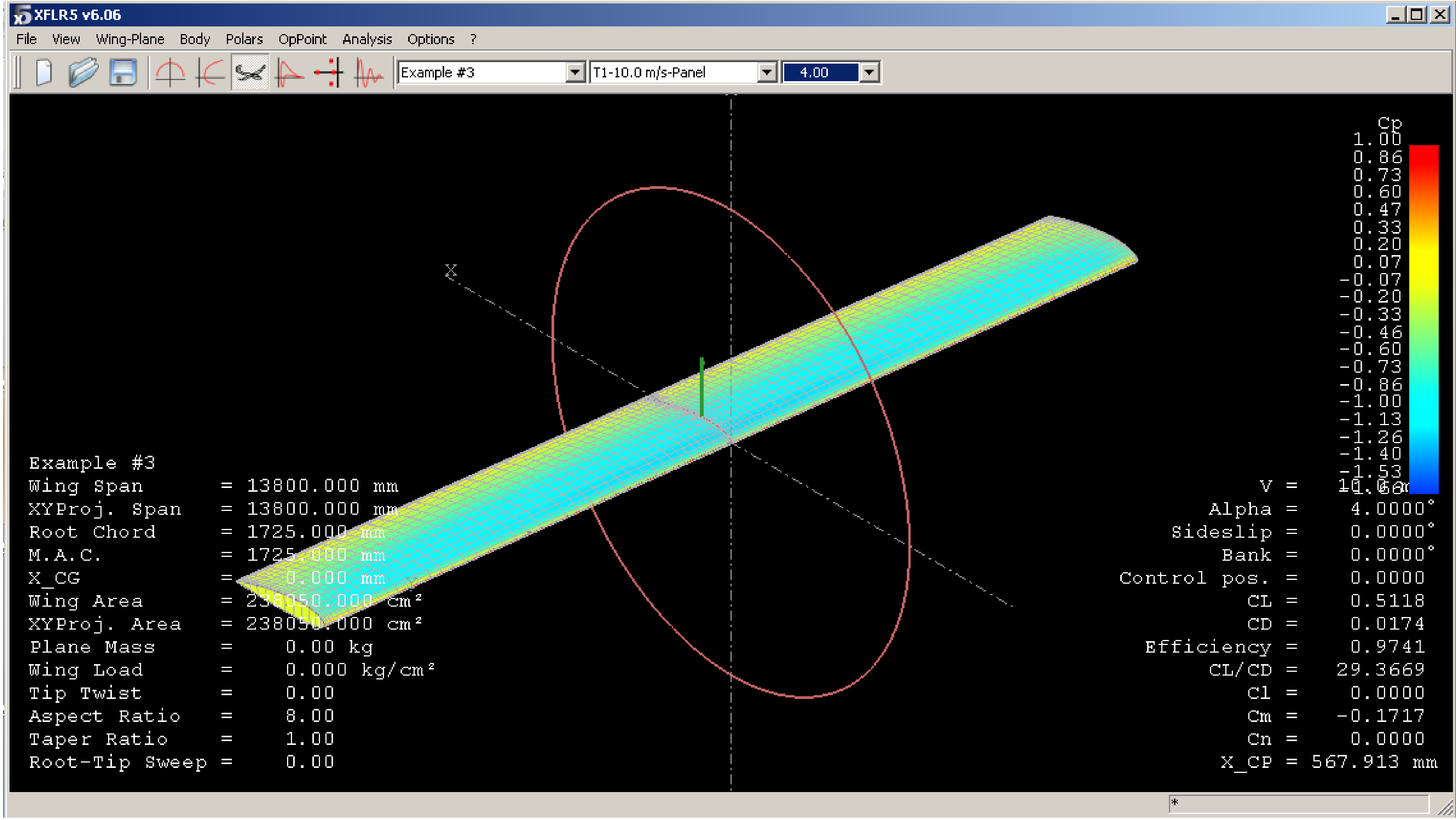


- Continuing from the LLT analysis
- Click <polars><define an analysis
- Choose 3D Panels
- Click OK
- Set angle attack range
- Click Analyze






XFLR5 v6.06
File View Wing-Plane Body Polars OpPoint Analysis Options ?
Example #3
T1-10.0 m/s-Panel
-5.00
CL
CD
Alpha
Cm
CL/CD
Example #3
T1-10.0 m/s-LLT
T1-10.0 m/s-Panel



Right Click on graph, choose current op point, export
Generates file of data with pressures on all panels