Appendix A: XFOIL Tutorial

In this section we will give a brief step-by-step explanation on how to use XFOIL to do the most common airfoil analysis tasks, including

- Read in airfoil data and define NACA airfoils
- Define and deflect a flap and evaluate flap hingemoments
- Perform inviscid and viscous calculations
- Save results to a file

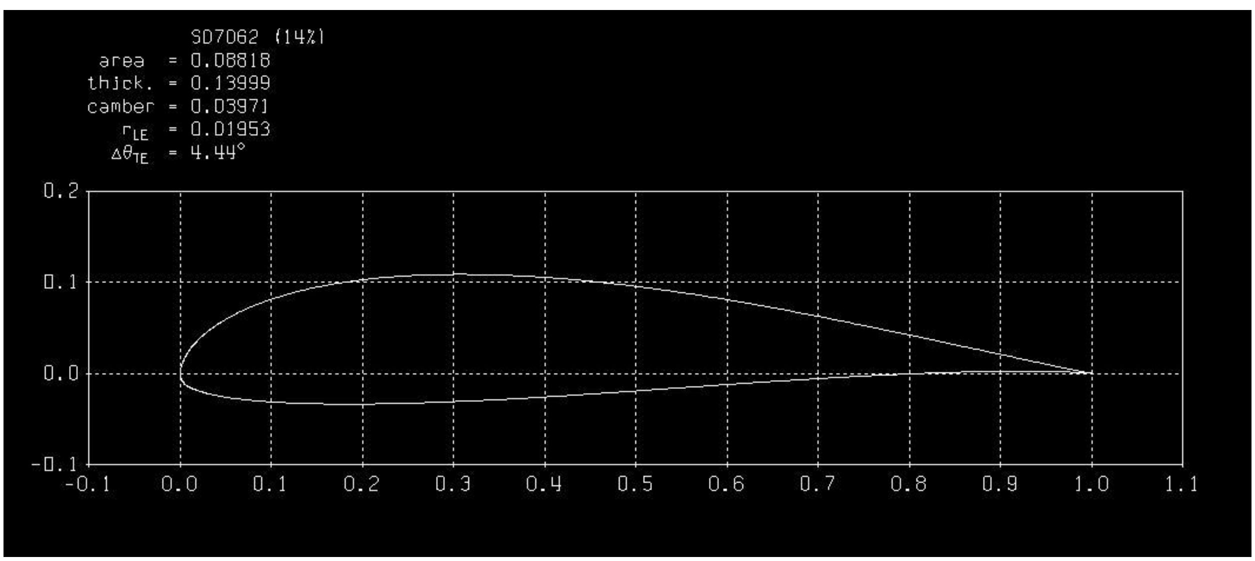
The opening screen is below. The items with a period in front of them involve going into a new menu. At any time typing "?" will bring up the local menu.

```
 ===================================================
  XFOIL Version 6.96
  Copyright (C) 2000   Mark Drela, Harold Youngren

  This software comes with ABSOLUTELY NO WARRANTY,
    subject to the GNU General Public License.

  Caveat computor
 ===================================================

 File  xfoil.def  not found

   QUIT    Exit program

  .OPER    Direct operating point(s)
  .MDES    Complex mapping design routine
  .QDES    Surface speed design routine
  .GDES    Geometry design routine

   SAVE f  Write airfoil to labeled coordinate file
   PSAV f  Write airfoil to plain coordinate file
   ISAV f  Write airfoil to ISES coordinate file
   MSAV f  Write airfoil to MSES coordinate file
   REVE    Reverse written-airfoil node ordering

   LOAD f  Read buffer airfoil from coordinate file
   NACA i  Set NACA 4,5-digit airfoil and buffer airfoil
   INTE    Set buffer airfoil by interpolating two airfoils
   NORM    Buffer airfoil normalization toggle
   XYCM rr Change CM reference location, currently  0.25000 0.00000

   BEND    Display structural properties of current airfoil

   PCOP    Set current-airfoil panel nodes directly from buffer airfoil points
   PANE    Set current-airfoil panel nodes ( 160 ) based on curvature
  .PPAR    Show/change paneling

  .PLOP    Plotting options

   WDEF f  Write  current-settings file
   RDEF f  Reread current-settings file
   NAME s  Specify new airfoil name
   NINC    Increment name version number

   Z       Zoom    | (available in all menus)
   U       Unzoom  |

 XFOIL   c>
```

The first step is to load an airfoil file. This is done by typing “load airfoil_name”, in the example “load sd7062.dat”. Alternatively it is possible to use “NACA 4412”, for example, to input NACA 4-series airfoils. The airfoil characteristics are displayed. The airfoil is 14% thick and has 4% camber. The “pane” command will re-panel the airfoil coordinates to have 160 instead of the original 61 nodes. Too few nodes will cause numerical problems later on; it is therefore recommended to use this command for every airfoil that has fewer than 160 points.

```
XFOIL   c>  load sd7062.dat

Labeled airfoil file.  Name:    SD7062 (14%)

Number of input coordinate points:  61
Counterclockwise ordering
Max thickness =     0.139994  at x =   0.272
Max camber    =     0.039706  at x =   0.388

 LE  x,y  =  -0.00002  -0.00006  |   Chord =   1.00003
 TE  x,y  =   1.00001   0.00000  |

Current airfoil nodes set from buffer airfoil nodes (  61 )

XFOIL   c>  pane

Sharp trailing edge

Paneling parameters used...
  Number of panel nodes       160
  Panel bunching parameter    1.000
  TE/LE panel density ratio   0.150
  Refined-area/LE panel density ratio    0.200
  Top    side refined area x/c limits  1.000 1.000
  Bottom side refined area x/c limits  1.000 1.000

XFOIL   c> _
```

If we open the GDES menu (by typing in”GDES”) the airfoil geometry shows up in a separate window. The airfoil defining characteristics are displayed along with a picture.



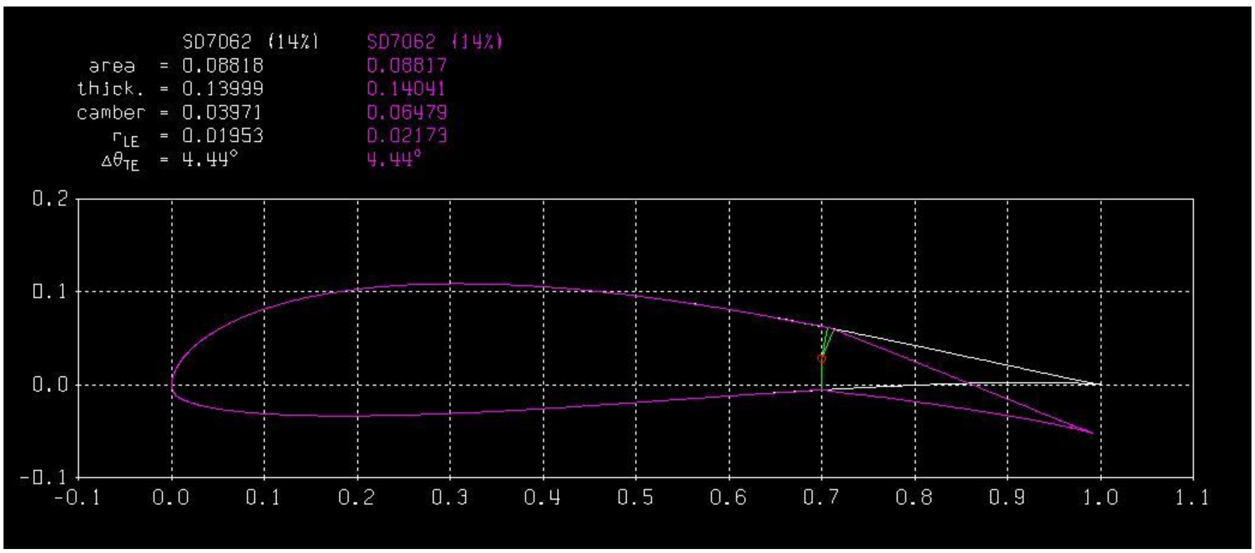
The GDES menu options are shown below. We will only use the flap commands, but it is possible to design and/or modify an airfoil.

```
.GDES   c>  ?

   <cr>     Return to Top Level
   !        Redo previous command

   GSET     Set buffer  airfoil <== current airfoil
   eXec     Set current airfoil <== buffer  airfoil
   SYMM     Toggle y-symmetry flag

   ADEG r   Rotate about origin (degrees)
   ARAD r   Rotate about origin (radians)
   Tran rr  Translate
   Scal r   Scale about origin
   LINS rr. Linearly-varying y scale
   DERO     Derotate (set chord line level)

   TGAP rr  Change trailing edge gap
   LERA rr  Change leading edge radius

   TCPL     Toggle thickness and camber plotting
   TFAC rr  Scale existing thickness and camber
   TSET rr  Set new thickness and camber
   HIGH rr  Move camber and thickness highpoints
  .CAMB     Modify camber shape directly or via loading

   Flap rrr Deflect trailing edge flap

   Modi     Modify contour via cursor
   SLOP     Toggle modified-contour slope matching flag

   CORN     Double point with cursor (set sharp corner)
   ADDP     Add    point with cursor or keyboard x,y
   MOVP     Move   point with cursor or keyboard x,y
   DELP     Delete point with cursor

   UNIT     Normalize buffer airfoil to unit chord
   Dist     Determine distance between 2 cursor points
   CLIS     List curvatures
   CPLO     Plot curvatures
   CANG     List panel corner angles
   CADD ri. Add points at corners exceeding angle threshold

   Plot     Replot buffer airfoil
   INPL     Replot buffer airfoil without scaling (in inches)
   Blow     Blowup plot region
   Rese     Reset plot scale and origin
   Wind     Plot window adjust via cursor and keys

   TSIZ r   Change tick-mark size
   TICK     Toggle node tick-mark plotting
   GRID     Toggle grid plotting
   GPAR     Toggle geometric parameter plotting
   Over f   Overlay disk file airfoil

   SIZE r   Change absolute plot-object size
  .ANNO     Annotate plot
   HARD     Hardcopy current plot

   NAME s   Specify new airfoil name
   NINC     Increment name version number

.GDES   c>
```

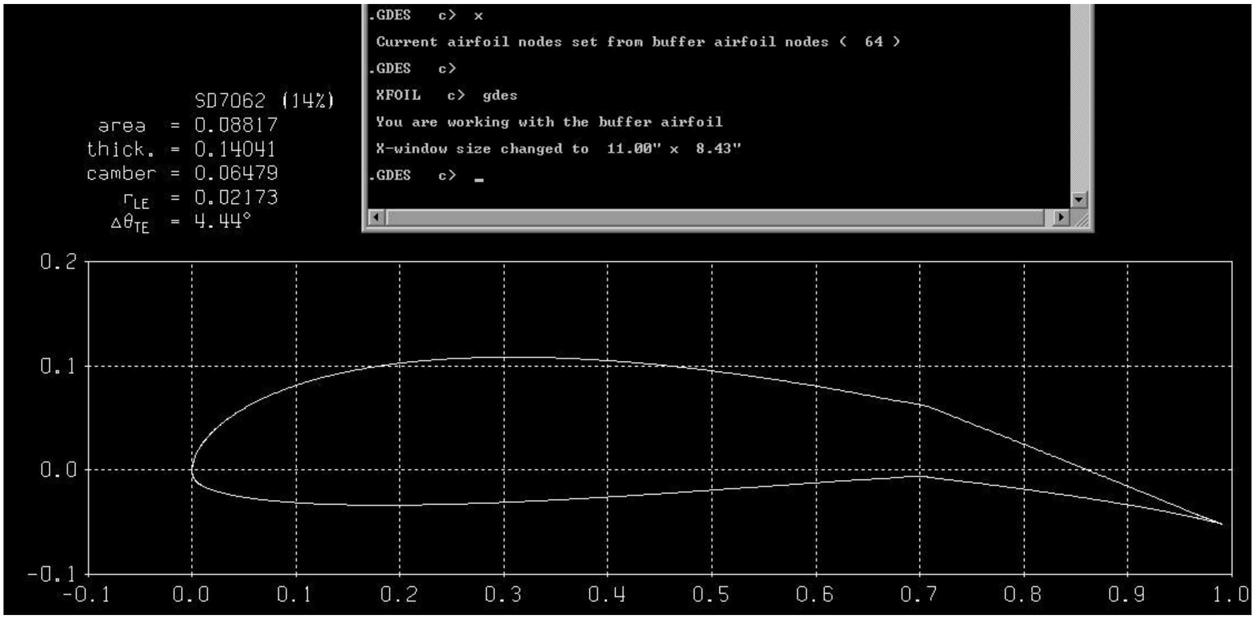
To define a 30% flap type in “flap”. The x-location of the hinge is one minus the flap-to-chord ratio. Instead of explicitly defining the hinge y location the 999 command allows the hinge location to be specified in terms of the thickness. In this case the hinge is located at half the thickness and is deflected 10 degrees.

```
.GDES   c>  flap

Enter flap hinge x location   r>  0.7

  Top    surface:  y =  0.0623      y/t = 1.0
  Bottom surface:  y = -0.0057      y/t = 0.0

Enter flap hinge y location (or 999 to specify y/t)   r>  999

Enter flap hinge relative y/t location   r>  0.5

 Flap hinge: x,y =  0.70000  0.02829

Enter flap deflection in degrees (+ down)   r>  10

 Top breaks: x,y =    0.70648  0.06105       0.70648  0.06105
 Bot breaks: x,y =    0.69905 -0.00580       0.70498 -0.00545
 Max thickness =     0.140413  at x =   0.279
 Max camber    =     0.064785  at x =   0.627

.GDES   c>  _
```

The geometry window now shows the new airfoil characteristics along with the old one.



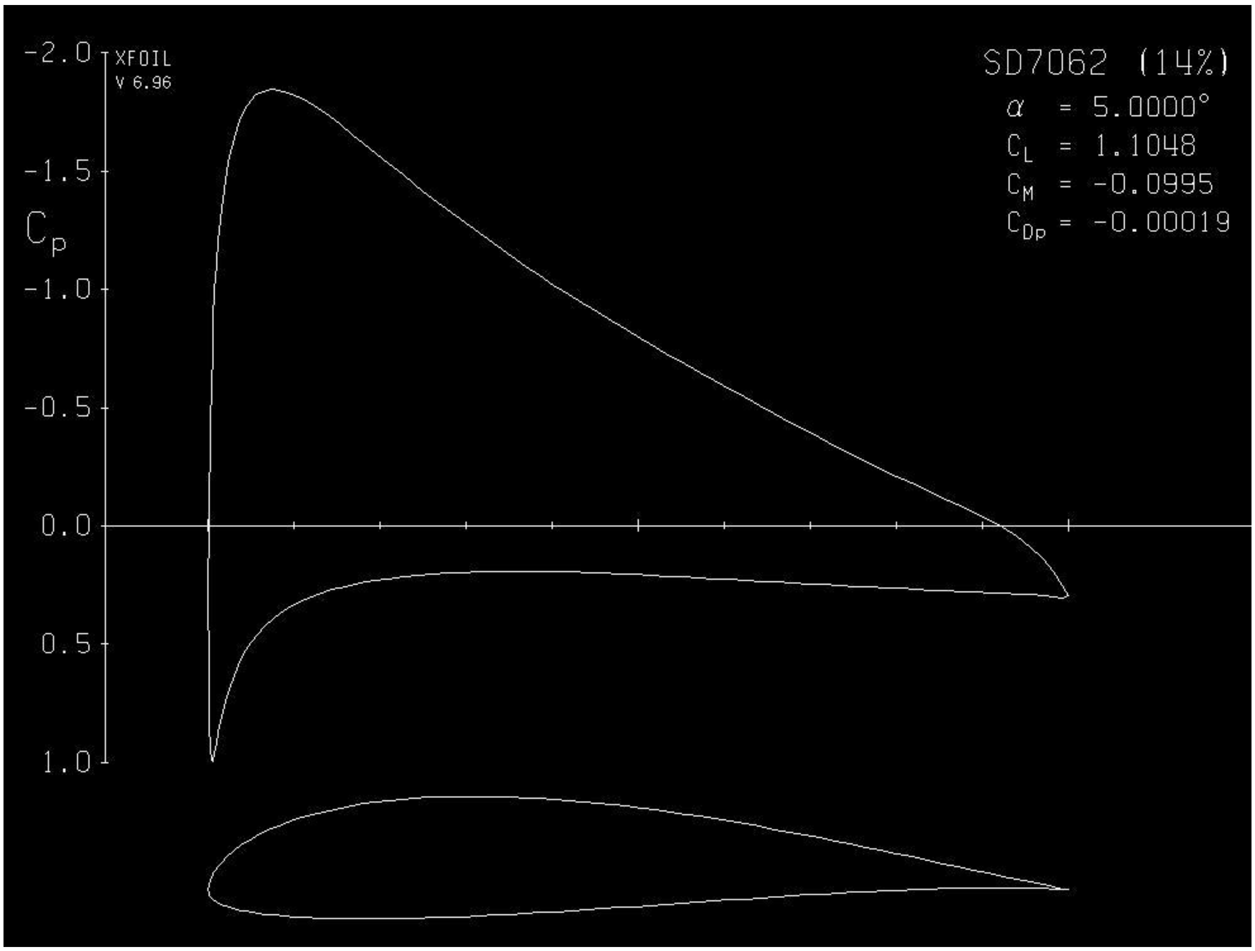
To actually use the flapped airfoil section it is necessary to use the "x" command to reset the airfoil to be the flapped one that we just defined. This eliminates the old airfoil and allows the flapped one to be used.



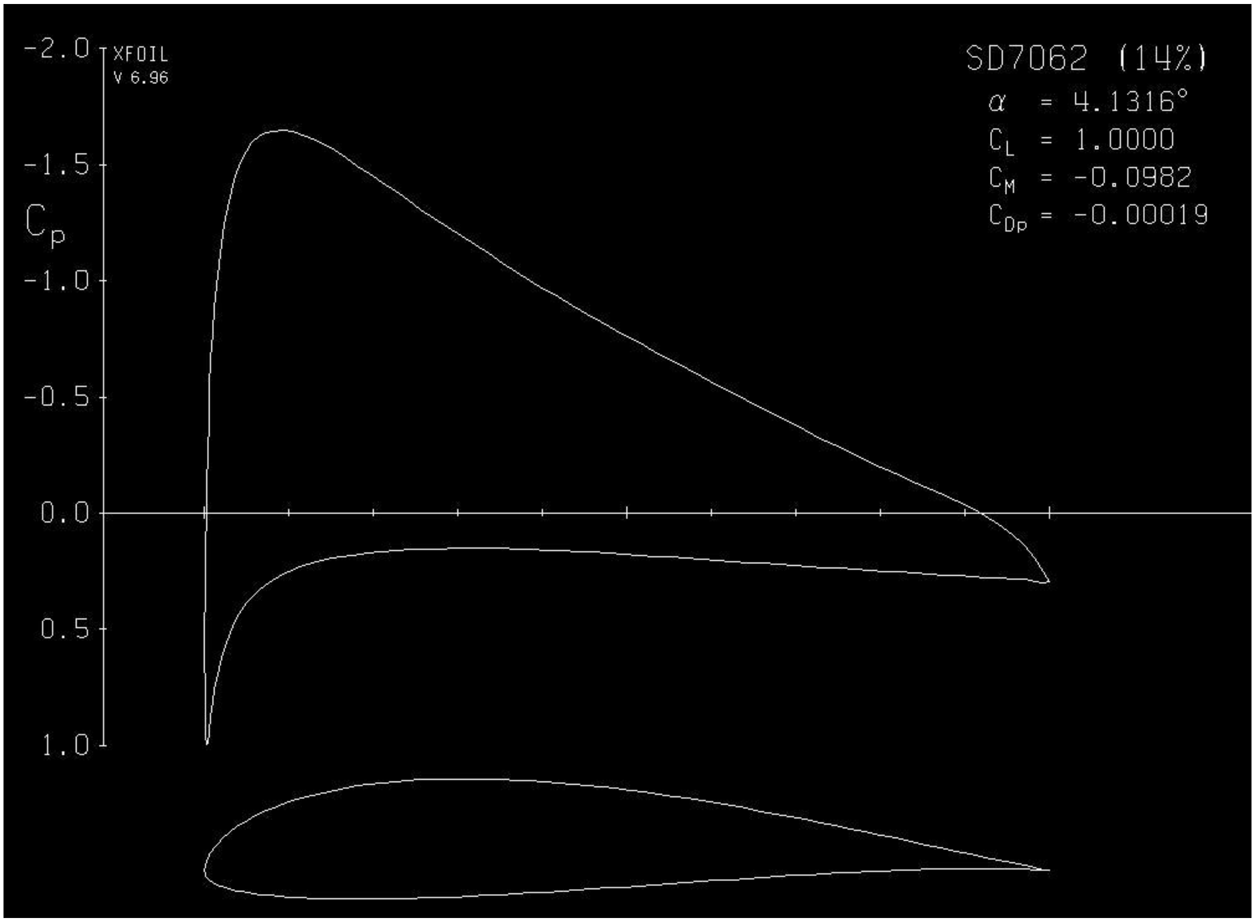
Now we can look at the OPER menu where the actual calculations are done. When first opened the OPER mode is set to inviscid. The inviscid calculations are extremely fast, but only marginally accurate for lift and not at all for drag.

```
.GDES   c>

 XFOIL   c>  oper

.OPERi   c>  ?

   <cr>      Return to Top Level
   !         Redo last ALFA,CLI,CL,ASEQ,CSEQ,VELS

   Visc r    Toggle Inviscid/Viscous mode
  .VPAR      Change BL parameter(s)
   Re   r    Change Reynolds number
   Mach r    Change Mach number
   Type i    Change type of Mach,Re variation with CL
   ITER      Change viscous-solution iteration limit
   INIT      Toggle BL initialization flag

   Alfa r    Prescribe alpha
   CLI  r    Prescribe inviscid CL
   Cl   r    Prescribe CL
   ASeq rrr  Prescribe a sequence of alphas
   CSeq rrr  Prescribe a sequence of CLs

   SEQP      Toggle polar/Cp(x) sequence plot display
   CINC      Toggle  minimum Cp  inclusion in polar
   HINC      Toggle hinge moment inclusion in polar
   Pacc i    Toggle auto point accumulation to active polar
   PGET f    Read new polar from save file
   PWRT i    Write polar to save file
   PSUM      Show summary of stored polars
   PLIS i    List stored polar(s)
   PDEL i    Delete stored polar
   PSOR i    Sort stored polar
   PPlo ii.  Plot stored polar(s)
   APlo ii.  Plot stored airfoil(s) for each polar
   ASET i    Copy stored airfoil into current airfoil
   PREM ir.  Remove point(s) from stored polar
   PNAM i    Change airfoil name of stored polar
   PPAX      Change polar plot axis limits

   RGET f    Read new reference polar from file
   RDEL i    Delete stored reference polar

   GRID      Toggle Cp vs x grid overlay
   CREF      Toggle reference Cp data overlay
   FREF      Toggle reference CL,CD.. data display

   CPx       Plot Cp vs x
   CPV       Plot airfoil with pressure vectors (gee wiz)
  .VPlo      BL variable plots
  .ANNO      Annotate current plot
   HARD      Hardcopy current plot
   SIZE r    Change plot-object size
   CPMI r    Change minimum Cp axis annotation

   BL   i    Plot boundary layer velocity profiles
   BLC       Plot boundary layer velocity profiles at cursor
   BLWT r    Change velocity profile scale weight

   FMOM      Calculate flap hinge moment and forces
   FNEW rr   Set new flap hinge point
   VELS rr   Calculate velocity components at a point
   DUMP f    Output Ue,Dstar,Theta,Cf vs s,x,y to file
   CPWR f    Output x vs Cp to file
   CPMN      Report minimum surface Cp
   NAME s    Specify new airfoil name
   NINC      Increment name version number

.OPERi   c>
```

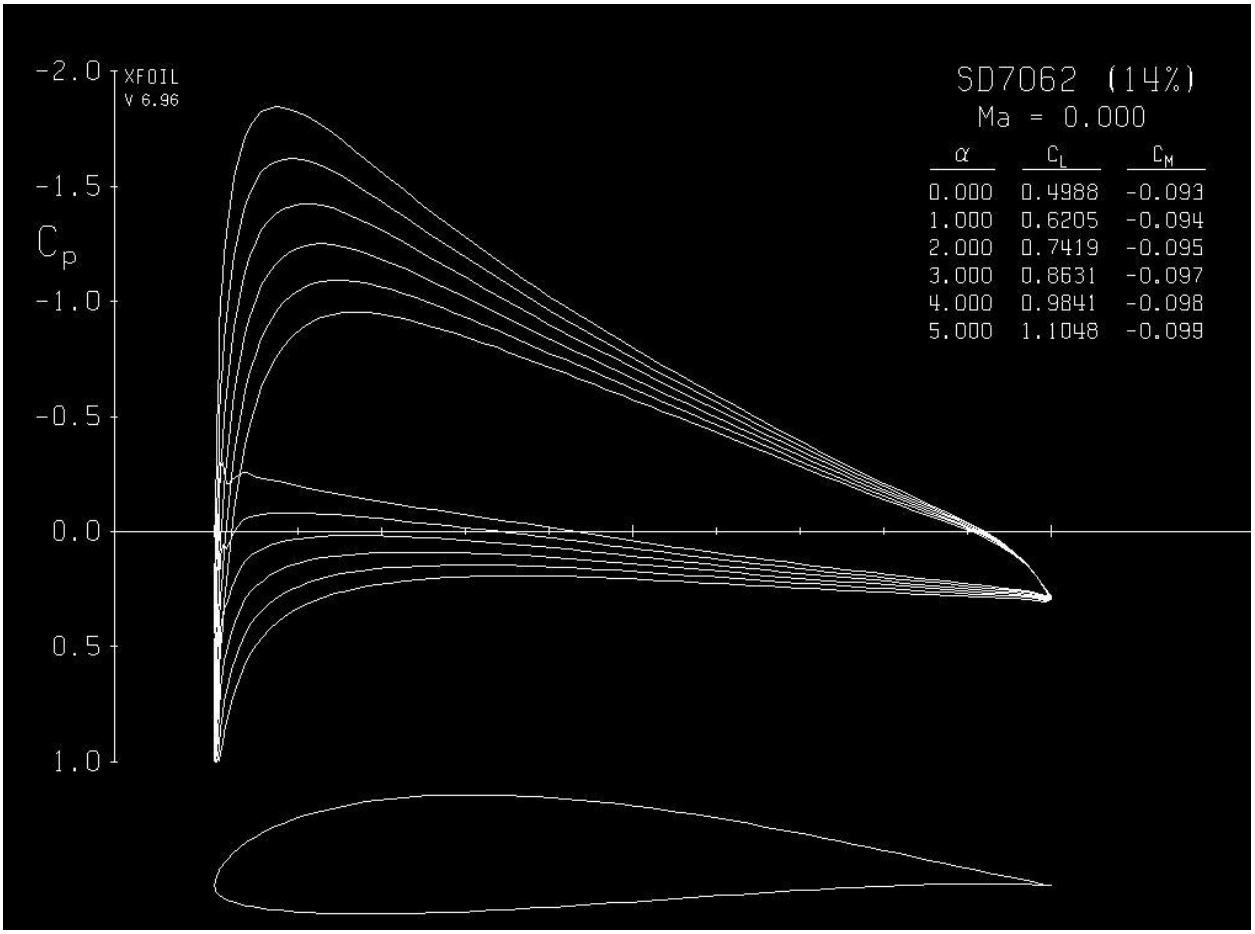
By typing in “alfa 5” we can get the airfoil characteristics at 5 degree angle of attack. The Cp plot is shown as well as the lift, drag and pitching moment characteristics. Because we are solving the inviscid solution the drag is essentially zero. The Cp plot shows the variation in pressure coefficient over the chord for both the upper and lower surfaces. Note that negative Cp is up. The difference between the upper surface and lower surface Cp is equal to the lift produced.



We can also set the lift coefficient and have XFOIL find the angle of attack by specifying “cl 1”. The angle of attack in this case is 4.1316 degrees.



We are not limited to one angle of attack or lift coefficient at a time. We can also go through a sequence of alphas using the "aseq" command. An example is "as 0 5 1". This will go through 0, 1, 2, 3, 4 and 5 degrees alpha. The notation is command name, start alpha, end alpha, increment. The resulting plot shows all the CP diagrams, as well as the lift and pitching moment for each alpha.



It is also possible to save a batch of results to a text file. The easiest way to do this is the "pacc" command. From here the user can set the polar save filename; it is also possible but not necessary to save the dump results. Once "pacc" is enable all of the results will be saved until pacc is input again to turn off the saving mode. In the following we save the results to "temp_polar.dat" for a simple run.

```
.OPERi   c>  as 0 5 1

.OPERi   c>  pacc

 Polar  1 newly created for accumulation
 Airfoil archived with polar: SD7062 (14%)

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

 New polar save file available

Enter  polar dump filename  OR  <return> for no file   s>

 Polar dump file will NOT be written

 Polar accumulation enabled

.OPERia   c>  as 0 10 1

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

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 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

 Point added to stored polar  1
 Point written to save file  temp_polar.dat
 Dump file unspecified or not available

.OPERia   c> _
```
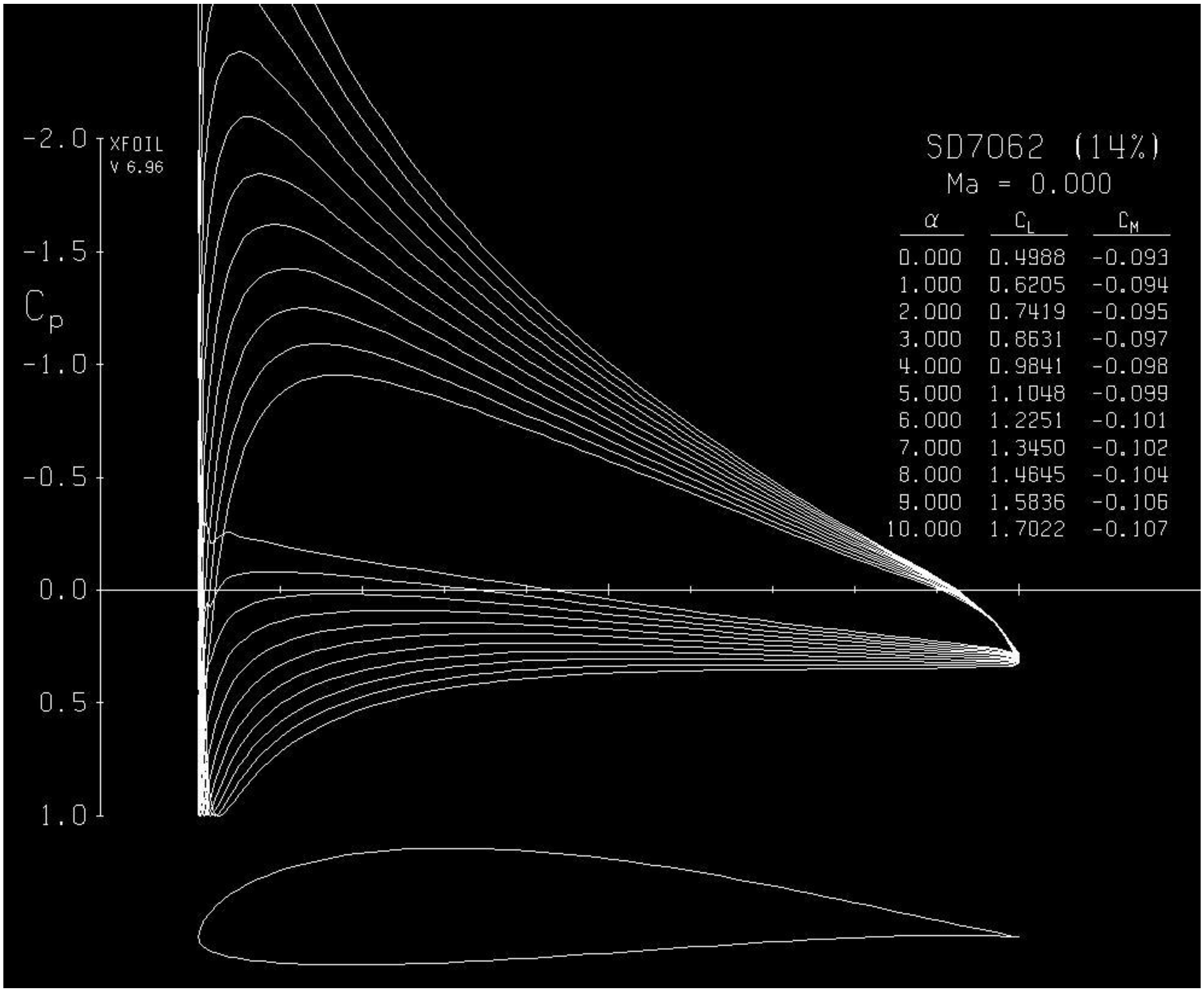

This is what the other window displays after the sequence. The data in the upper right are written to the file. Note that the Cp diagram is truncated. It is possible to change the limits on the graph, but not necessary.



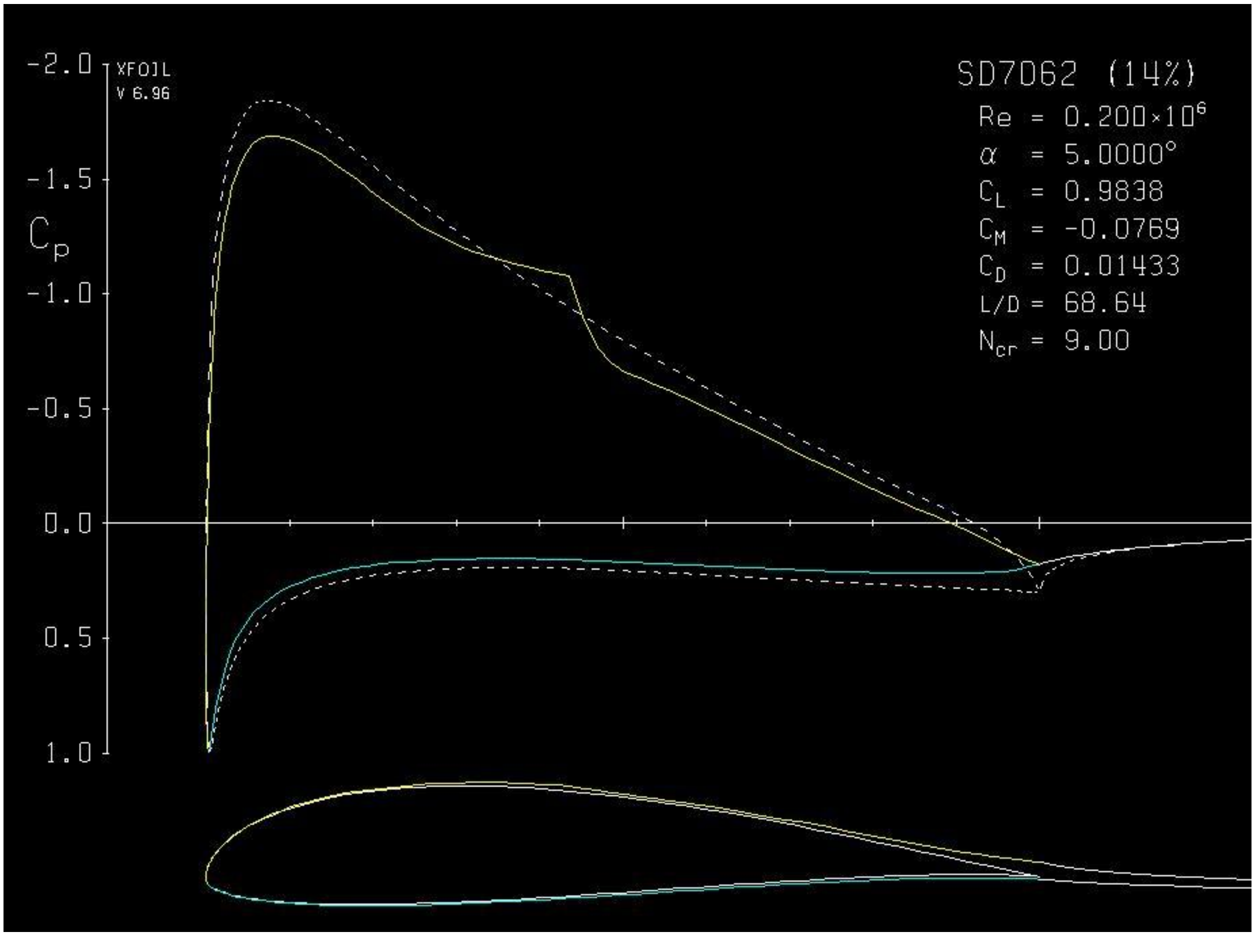
Now we can move to viscous solutions using the “visc” command. This command prompts for a Reynolds number. The OPERi menu then becomes the OPERv menu to designate the change in solution method. By typing in “re” it is possible to change the Reynolds Number input. Typing “visc” again will turn off the viscous mode and go back to inviscid solutions.

```
.OPERi   c>  visc

Enter Reynolds number   r>  3e5

 M  =     0.0000
 Re =     300000

.OPERv   c>  re

 Currently...

 Re =     300000

Enter new Reynolds number   r>  2e5

.OPERv   c>  _
```

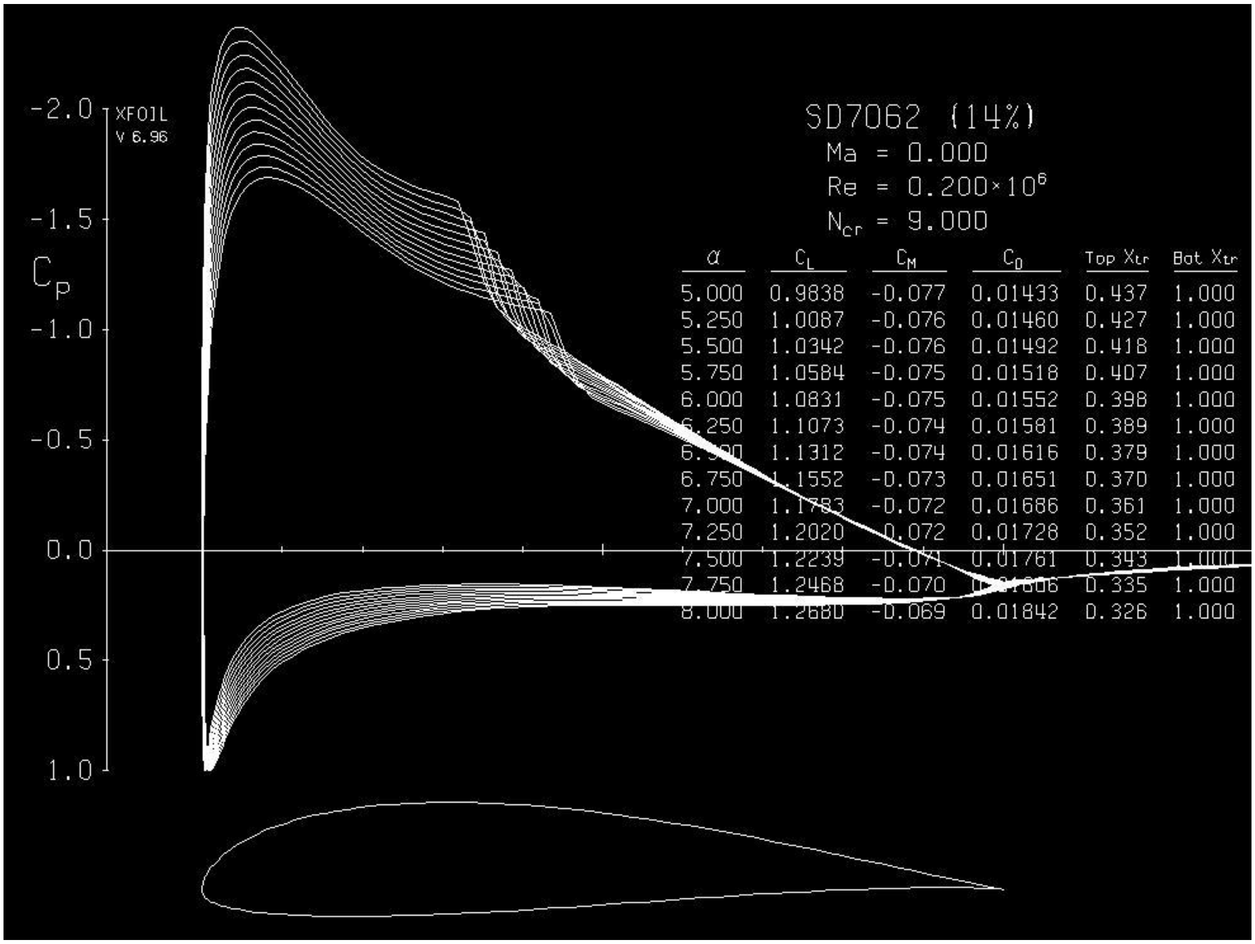
When we type in “alfa 5” now things are much more interesting. The below shows the last couple of iterations. In this case the convergence failed. When that occurs type in “!” to continue iterating. The default is for a mere 10 iterations. It is recommended that this be increased to 100 to avoid failed iterations that will be quite annoying when we start sequencing in alpha to create drag polars. The command to do this is “iter 100”.

```
  9   rms: 0.7496E-02   max: -.5758E-01   C at   73  2
      a =  5.000      CL =  0.9837
     Cm = -0.0769     CD =  0.01433    =>   CDf =  0.00758    CDp =  0.00675

 Side 1  free  transition at x/c =  0.4373   56
 Side 2 forced transition at x/c =  1.0000   73

 10   rms: 0.3723E-03   max: 0.2787E-02   D at    4  2
      a =  5.000      CL =  0.9838
     Cm = -0.0769     CD =  0.01433    =>   CDf =  0.00752    CDp =  0.00682
 VISCAL:  Convergence failed
 Type "!" to continue iterating

.OPERv   c>  !

 Solving BL system ...

 Side 1  free  transition at x/c =  0.4372   56
 Side 2 forced transition at x/c =  1.0000   73

  1   rms: 0.2033E-05   max: -.1637E-04   D at   58  1
      a =  5.000      CL =  0.9838
     Cm = -0.0769     CD =  0.01433    =>   CDf =  0.00752    CDp =  0.00681

 X-window size changed to  11.00" x  8.20"

.OPERv   c>  iter 100

.OPERv   c>
```

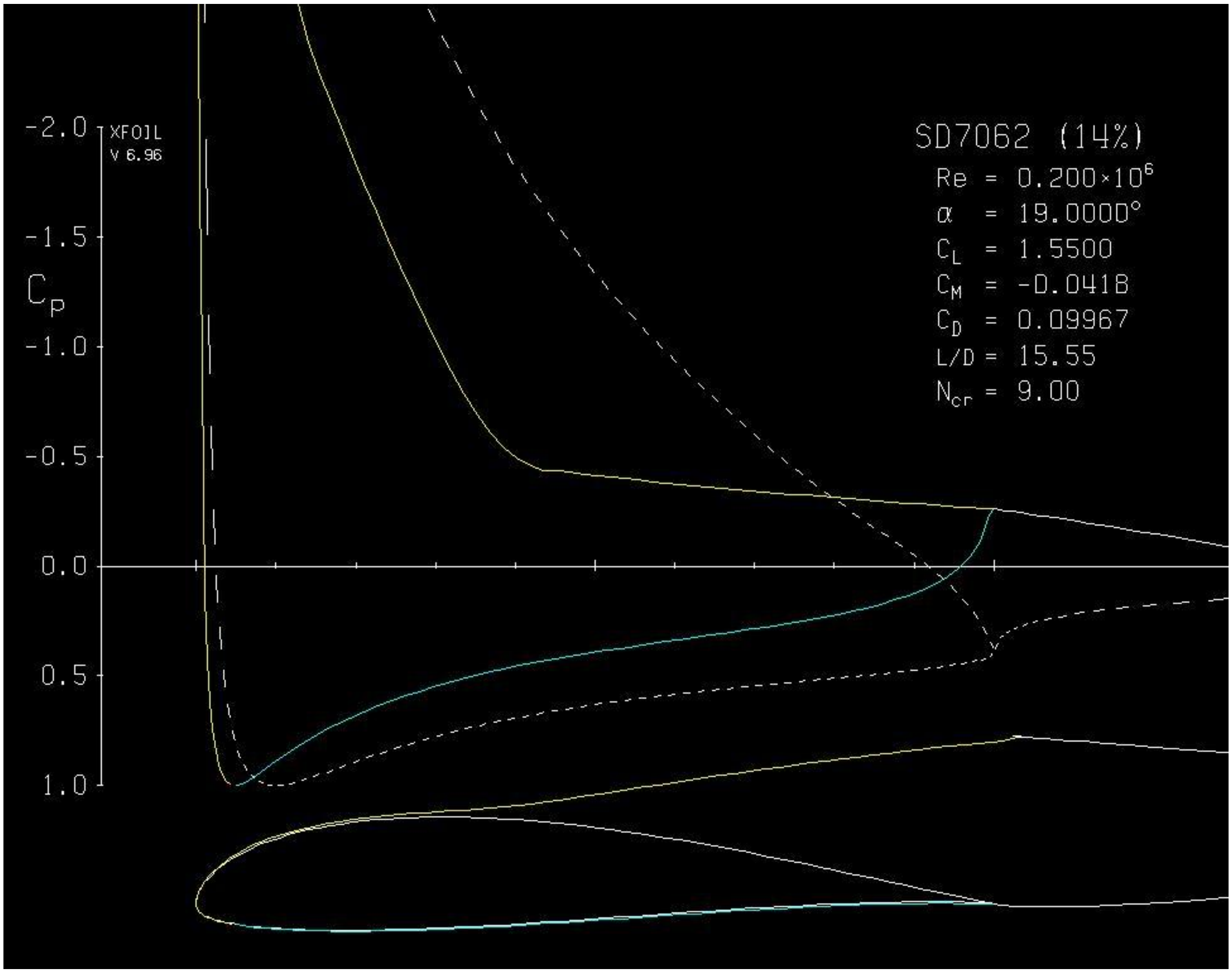
The CP diagram and results now look different as well. The inviscid solution is still shown in dashed form. The airfoil shape as modified to account for the boundary layer is also explicitly shown. The drag data is now reasonable and the lift-to-drag ratio is displayed along with the Reynolds Number.



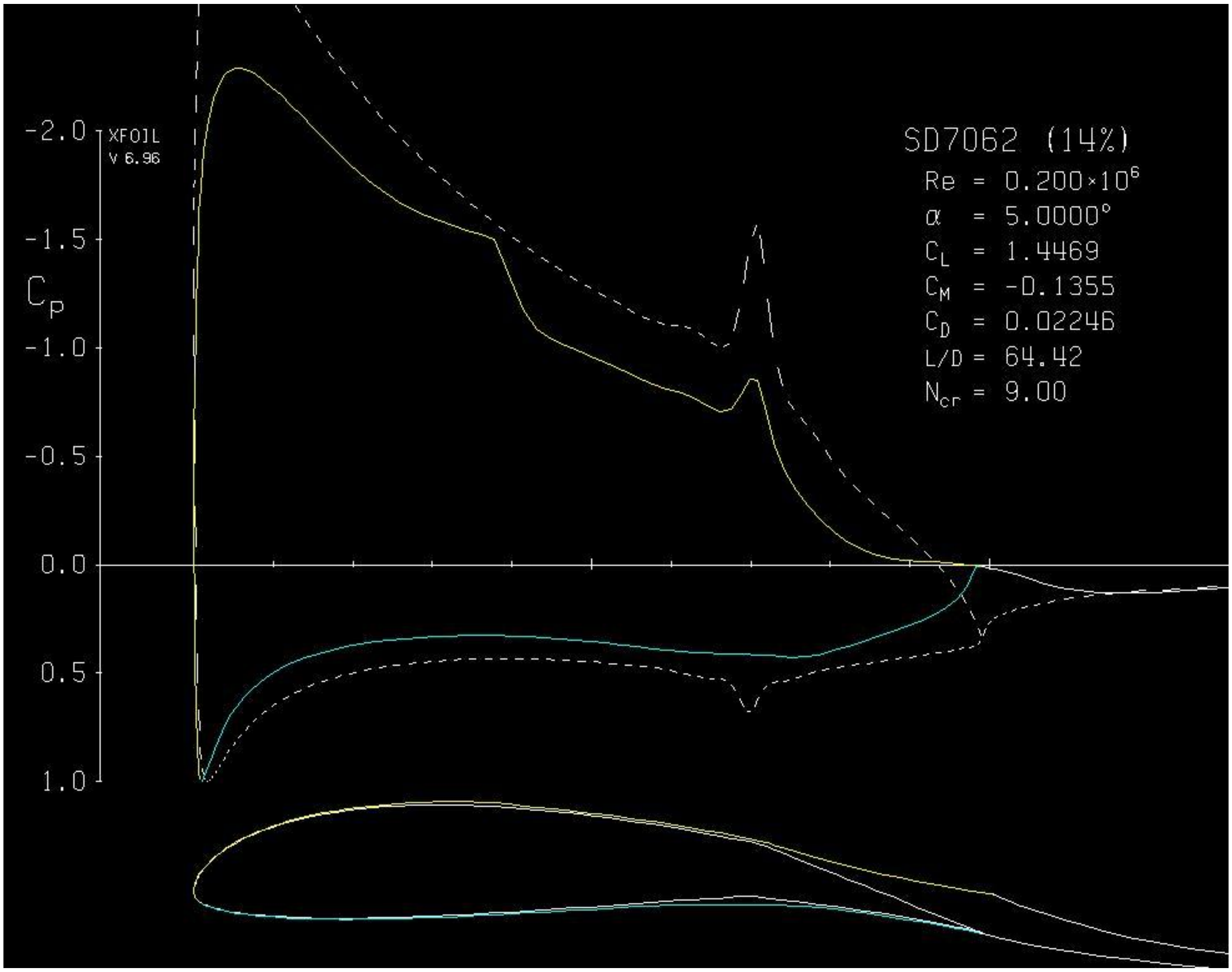
The sequence command can again be used to go through a series of alphas. This can again be used with the pacc command to create a drag polar. With the viscous solution it is highly recommended that very small angle of attack steps be taken as the airfoil nears stall. It is possible to get reasonable results up to and even a little past stall, but it must be done carefully. This is where increasing the iterations to 100 (or more) will pay off in not having constant convergence failures.



An example plot of what happens to the viscous solution around stall is shown below. The airfoil shape is now bumped up to include a large portion of wake where viscous effects dominate.



If we go back and redefine the flapped airfoil we can also get viscous results. There is a significant discontinuity in the Cp caused by the flap transition.



The last thing we would like to do is extract flap hingemoments. This is done by invoking "fmom" from the OPER menu. This provides the hingemoment per span as in the following. To evaluate the actual hingemoment only requires knowing the flight condition, the local chord and the span of the flap.

```
.OPERv   c>  fmom

 Flap hinge x,y :  0.7000  0.0283
                                             2  2
 Hinge moment/span = 0.014613  x  1/2 rho V   c
                                             2
 x-Force     /span = 0.019805  x  1/2 rho V   c
                                             2
 y-Force     /span = 0.156429  x  1/2 rho V   c

.OPERv   c>
```

Thus concludes our tutorial. It is by no means intended to be complete. The user can read the user guide for a bit more explanation, but not all commands are explained there. The best method is to experiment and play around with the program. This should give the reader a solid base from which to perform such experimentation.

To summarize here are the key commands needed to go through an example session where we load an airfoil file, modify it to have 30% flaps deflected 10 degrees, and record a drag polar from 0 to 10 degrees with the viscous solution at 200,000 Reynolds Number. Empty lines require an extra enter, and explanations of each command are on the right.

Command	Explanation
load sd7062.dat	load the airfoil
pane	re-panel to 160 nodes
gdes	go to GDES menu
flap	define a flap
0.7	30% flap to chord
999	specify hinge in terms of thickness
0.5	hinge halfway up thickness
10	deflected ten degrees
x	needed to re-set airfoil
	go back to main menu
oper	go to OPER menu
visc	use viscous solution
2e5	200,000 Reynolds Number
pacc	turn on polar accumulation
temp_polar.dat	filename to save in
	don’t save dump file
as 0 10 1	go through sequence of alphas
pacc	turn off polar accumulation
	back to main menu
quit	exit XFOIL