Airfoil Tools

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Lift & drag polars

Many of the airfoils have polar diagrams which can be viewed in the details and comparison section sections of the site. These show the change in lift coefficient (Cl), drag coefficient (Cd) and pitching moment (Cm) with angle of attack (alpha). There is also a graph of lift coefficient (Cl) against drag coefficient (Cd) which gives the theoretical glide angle of the airfoil.

Xfoil

All the polar diagrams currently available have been produced using Xfoil, an application created by Mark Drela and Harold Youngren for the design and analysis of subsonic airfoils. There is more information and details on the limitations of the analysis on the **Xfoil page**.

I wanted to create polar diagrams for all the airfoils in the database for a range of Reynolds numbers and ncrit values that would typically be applicable for model or light aircraft or small wind turbines. The polars are keyed by three values.

- Reynolds number - These range from 50,000 to 1,000,000 in approximatly logarithmic steps. The Reynolds number is a dimensionless value that depends on the velocity, wing chord and fluid. There is more information on the **Reynolds number calculator** page.
- Ncrit value - This is used to model of the turbulence of the fluid or roughness of the airfoil. The **Xfoil documentation** has a section explaining the method and also the table (reproduced below) of suggested values:-

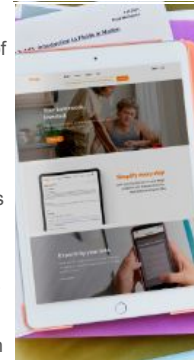| Situation	Ncrit
sailplane	12 to 14
motorglider	11 to 13
clean wind tunnel	10 to 12
average wind tunnel	9
dirty wind tunnel	4 to 8

- Mach number - This has been left at the default value of zero.

The polars were initially generated for an angle of attack (alpha) range -20° to +20° at 0.25° intervals. Some points never converge so this range may not be a continuous. The Xfoil lift and drag predictions are considered valid just beyond maximum lift (Cl max). Depending on the shape of the airfoil and Reynolds number this range may be some way past Cl max. After generating the polar files they were scanned to detect the maximum and minimum Cl values and truncated just past these values. Please be aware that most of this process was scripted so the cut off points may not be ideal. If too few points remain after this process then the polar was not included.

Xfoil has a command line user interface and this was automated using an **expect script** running under **Cygwin** on a Windows PC. The airfoil dat files are run through the parser to convert them to a Selig format and remove any extra text so they could be read by Xfoil. The commands to load the dat file and set up the session was fairly straight forward but it was more difficult to handle the user interaction when the iterative calculations failed to converge and restart Xfoil when it froze.

The commands for a typical scripted Xfoil session to generate a polar are below. The xfoil ALFA command was used for each individual angle of attack rather than the ASEQ as it was easier to automate, determine the succcess of the last calculation calculation and "step over" any failures. The polars were generated stating at ALFA = 0 in first a positive and then a negative direction. If a solution was found the next ALFA value was immediatley calculated. If no solution was found an INIT command was run before moving on to the next ALFA command.

```
LOAD dat/e1211-il.dat        Load the dat file
MDES                         Go to the MDES menu
FILT                         Smooth any variations in the dat file data
EXEC                         Execute the smoothing
                             Back to main menu
PANE                         Set the number and location of the airfoil points for analysis
OPER                         Go to the OPER menu
ITER 70                      Max number of iterations set to 70 for convergence
RE 50000                     Set Reynolds number (required?)
VISC 50000                   Set viscous calculation with Reynolds number
PACC                         Start polar output file
polar/e1211-il_50000.txt     The output polar file name
                             No dump file
ALFA 0                       Calculate lift and drag at 0° angle of attack
ALFA 0.25                    ... 0.25°
ALFA 0.5                     ... 0.5° ...
...                          ...more alpha calculations here ...
ALFA 3.5                     At 3.5° no convergence
ALFA 3.5                     ... try again ...
ALFA 3.5                     ... and again
INIT                         Run INIT to reinitialise
ALFA 3.75                    Skip to 3.75°
...                          ...rest of alpha calculations here ...
PACC                         Close polar file
VISC                         Reinitialise viscous calculation (required?)
                             Down to main menu
QUIT                         Exit Xfoil
```