Airfoil Write Up

Weston Pace

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1 Dictionary

Chord Line = the imaginary straight line joining the front edge and back edge of an airfoil

Camber = this shows the assymetry of the airfoil about the chord line (takes the average height above or belove the axis with respect to the upper and lower components of the airfoil)

Thickness = the distance between the top and lower points of the airfoil

Vortex = A mass of whirling fluid

Freestream Velocity = the velocity of the fluid far ahead of the object before it is affected by the object passing through it

Doublet = A source plus a sink

Coefficient of Drag = the ratio of drag on an object with respect to its velocity and surface area: $c_d = \frac{2F_d}{\rho A_d V_\infty^2}$

Coefficient of Lift = the ratio of lift on an object with respect to its velocity and surface area: $c_d = \frac{2F_l}{\rho A_l V_\infty^2}$

Pitching Moment Coefficient: Relates the ratio of the moment caused by the Lift force (and maybe drag force?) and the wing area, chord length, and the dynamic pressure: $C_m = \frac{M}{qAc}$

Angle of Attack = The angle formed between a bodies reference line and oncoming flow: α

Airfoil Polar = Sometimes a table or graph that shows the relationship between important aerodynamic constants: such as drag and lift coefficients

Lift Curve Slope = As angle of attack changes, the lift (coefficient of lift) changes with a theoretical slope of 2π (although is is usually a little lower)

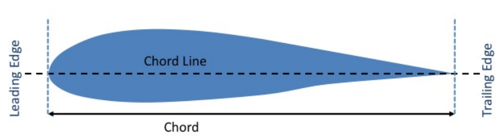

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Figure 1: Chord Line

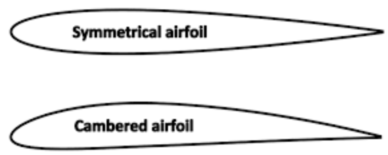

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Figure 2: Cambered vs Symmetric Airfoil

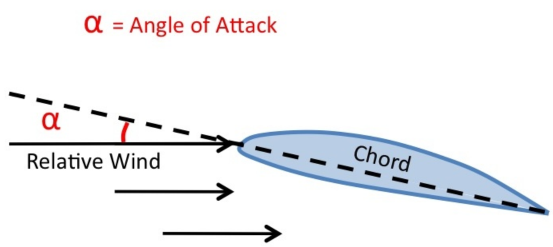

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Figure 3: Angle of Attack

Stall = When angle of attack is increased too high past the point of maximum lift, leading to a decrease in lift

Reynold's Number = The ratio of inertial forces to viscous forces of a fluid: $Re = \frac{\rho u L}{\mu}$ (u is the flow speed; L is the characteristic linear dimension; μ is the dynamic viscosity of the fluid)

Mach Number = The ratio of flow velocity past a boundary to the local speed of sound. More simply, it is a value that shows how many times faster an object is going than the speed of sound: M

Angle of Attack Direction: Wind coming from the left and airfoil is clockwise is positive alpha.

2 Methods for Finding Values of Xfoil

The important method for finding the values that Xfoil gives us stems from Thin Airfoil Theory. Thin Airfoil Theory is directly linked to Potential Flow theory, so for the solutions in Thin Airfoil Theory, we try to decide what would cause the flow around the airfoil to be tangent about the boundary surface of said airfoil. This is accomplished by placing an infinite line of point vortices and sources along the chord line of the airfoil. These vortices and sources induce a velocity on the flow about the boundary; however, the strength of each these vortices and sources is unknown with respect to their position on the chord line, so an analytical approach is needed to find functions that express this desired value. Once these functions are found, the velocity of the flow at each point around the boundary is also discovered. This allows for the pressure at each point to be known, and consequentially, this allows for the coefficients of lift, drag, and moment to be found.

Now, the solution method that Xfoil uses is similar to the Thin Airfoil approach, but it differs by how the airfoil is broken up and where we place the point vortices and sources. In Thin Airfoil Theory, the boundary of the airfoil can be described as a continuous line. However, Xfoil uses something called panel method, which, like the name implies, breaks the boundary of the airfoil into evenly sized panels. Additionally, panel method has each of the point vortices and sources along the panels instead of the chord line.

3 Using Xfoil

Xfoil is an extremely useful and easy to use package that allows for the coefficient of lift, drag, and moment to be found for an inputted airfoil geometry. The general idea behind using the package is that x and y coordinates for an airfoil are given to Xfoil, and then the program automatically generates panels with respect to these points. The only other things that the user has to do is choose values for the angle of attack, choose a Reynold's Number, choose a mach value, and generate lists for the calculated values to be pushed into. The methods of code required to do these processes are very simple, and results are produced quickly.

4 Effect of Angle of Attack on Coefficients

Although different airfoil geometries will produce somewhat unique polar diagrams, they all generally adhere to a general shape based on which coefficient is being observed.

The lift coefficient has a linear region around the angles of attack of -5 and 5 degrees. This is common to every airfoil, and the slope of this region has been analytically calculated to be around 2π. The slope is not always exactly 2π, but it is usually pretty close to this value. Additionally, most lift polars have a stall region following the linear region that shows a decrease in the coefficient of lift as the angle of attack is increased.

Additionally, for a symmetric airfoil, there is a coefficient of lift equal to zero when alpha is equal to zero. However, for a cambered airfoil, there is a coefficient of lift value greater than zero when the angle of attack is zero.

Instead of having a linear region, the drag coefficient polar follows a parabolic trend as alpha is either increased or decreased. In other words, the minimum drag is found at an angle of attack of zero and increases whether the angle of attack is increased or decreases.

The coefficient of moment polar is not as consistent as the other two, but it does somewhat follow a pattern. If the angle of attack is very negative, then the coefficient of moment polar is normally at its maximum, and if the sign of the angle of attack is changed, the the coefficient will be at its minimum. The polar in between these extremes is relatively random and consists of linear regions that have their slopes changing back and forth positive to negative.

5 Xfoil Validation

The chosen airfoil for this validation was the Drela HT05 airfoil. Furthermore, the comparative data was obtained from airfoiltools.com.

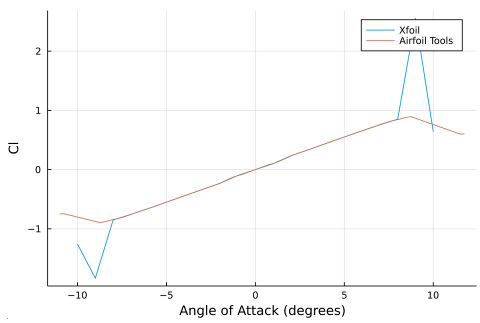


Figure 4: Coefficient of Lift Validation

Figure four shows that the data from Xfoil is similar to the data from airfoiltools.com. The linear region is almost indistinguishable from each other, and there is only slight difference in the regions where the solution begins to diverge.

Figure five also shows that in our expected convergence region, the airfoiltools.data and the Xfoil data are almost exactly the same.

Like the previous two, figure six shows that the Xfoil data is valid from around -10 to 10 degrees; however, there is a somewhat major divergence as we leave this region.

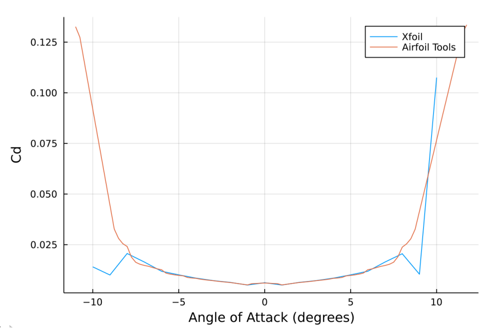
h 

Figure 5: Coefficient of Drag Validation

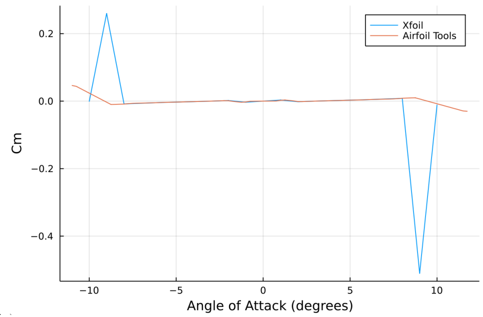


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Figure 6: Coefficient of Moment Validation

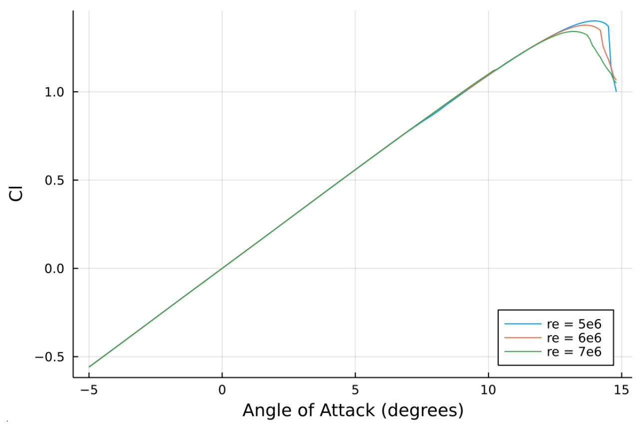

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Figure 7: Reynold's Number's Effect on Coefficient of Lift

6 Reynold's Number's Effect on Desired Outputs

Reynold's number is a non-dimensional value that can be expressed as $\frac{\rho u l}{\mu}$: where ρ is the density of the fluid, u is the flow speed, l is the length of the airfoil chord, and μ is the viscosity of the fluid. Reynold's number gives us a description of how the fluid will behave based off of the values that we put into it.

Adjusting the Reynold's number can play a huge part in the collected data. Figures 7-9 show the comparison of three different Reynold's number values on the airfoil coefficients. The airfoil used is still the Drela HT05.

Figure seven shows that as Reynold's number increases, stall angle decreases. This occurs because as Reynold's number increases, the momentum of the fluid increases: making it easier for the fluid to separate.

In figure eight, a similar trend to figure one appears. Although the polars

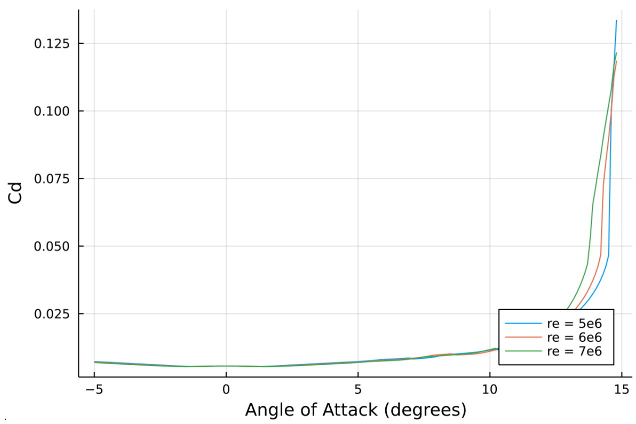

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Figure 8: Reynold's Number's Effect on Coefficient of Drag

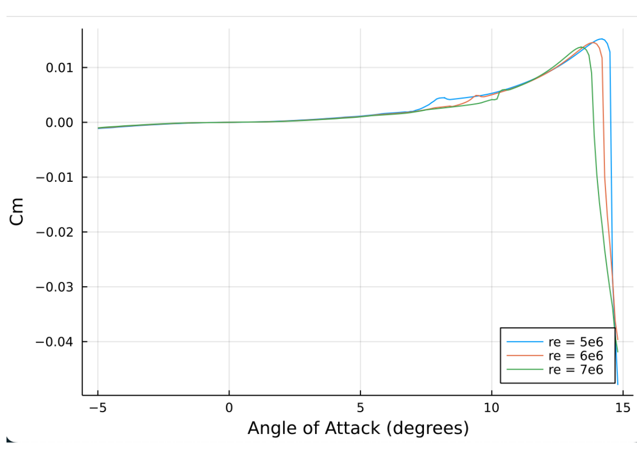

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Figure 9: Reynold's Number's Effect on Coefficient of Moment

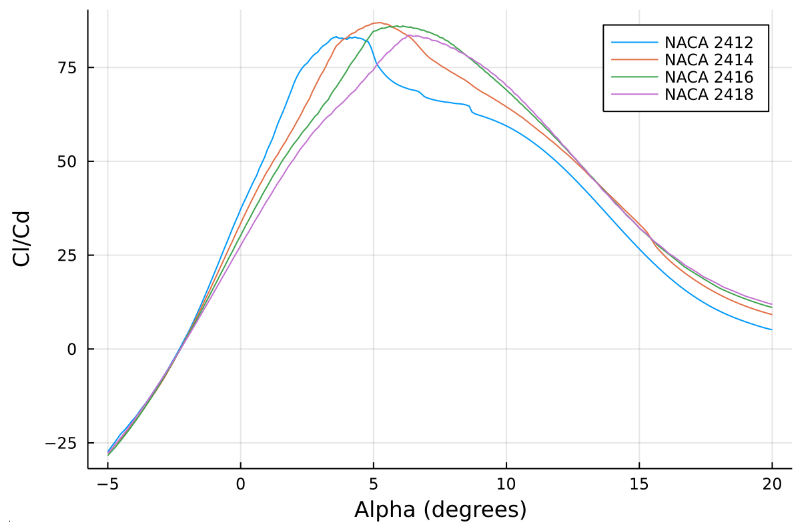

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Figure 10: Lift to Drag Ratio with Respect to Thickness

are very similar for lower angles of attack, a higher Reynold's number causes the coefficient of drag to increase sooner.

Similar to the last two comparisons, figure nine shows that as the Reynold's number increases, the coefficient of moment decreases at smaller angles of attack.

7 Effect of Camber and Thickness

In order to evaluate airfoils with different thicknesses and camber, I used the NACA airfoil.

Figure ten shows that thinner airfoils reach a maximum lift to drag ratio at a lower angler of attack.

The important thing that figure eleven shows is that in the linear region, slope of the lift curves for each thickness is around 2π. Additionally, we can see that thinner airfoils have a higher lift slope at lower angles of attack before stall.

The main takeaway from figure twelve is that a lower cambered airfoil has a lower maximum lift to drag ratio. Furthermore, at larger angle of attacks, a lower cambered airfoil has a higher lift to drag ratio than the higher cambered ones.

Figure thirteen has a lot going on, but an apparent trend is that in the linear region, the lift slope is around 2π even when the camber changes.

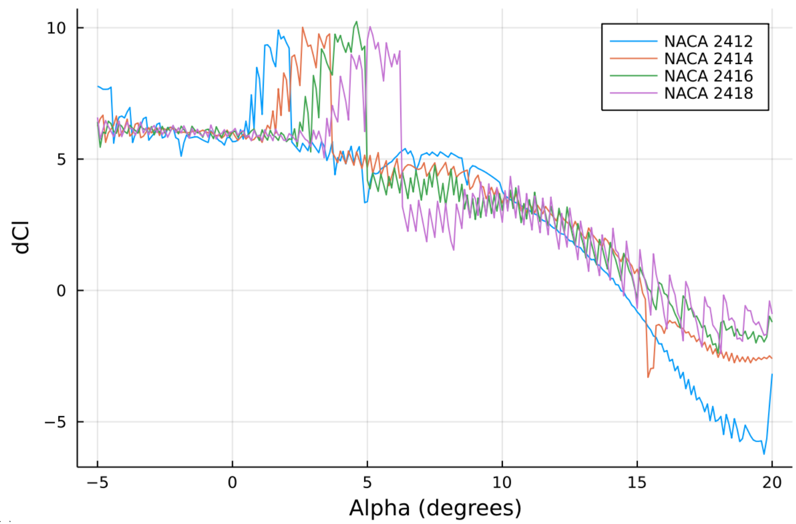

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Figure 11: Lift Derivative with Respect to Thickness

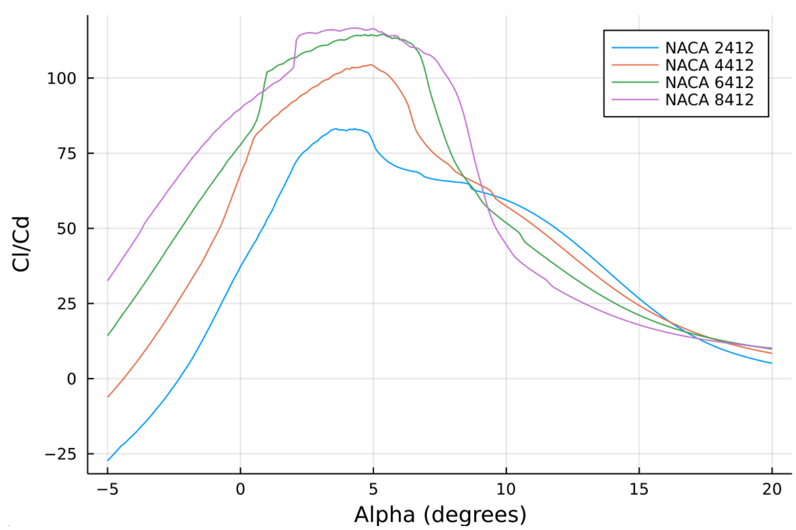

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Figure 12: Lift to Drag Ratio with Respect to Camber

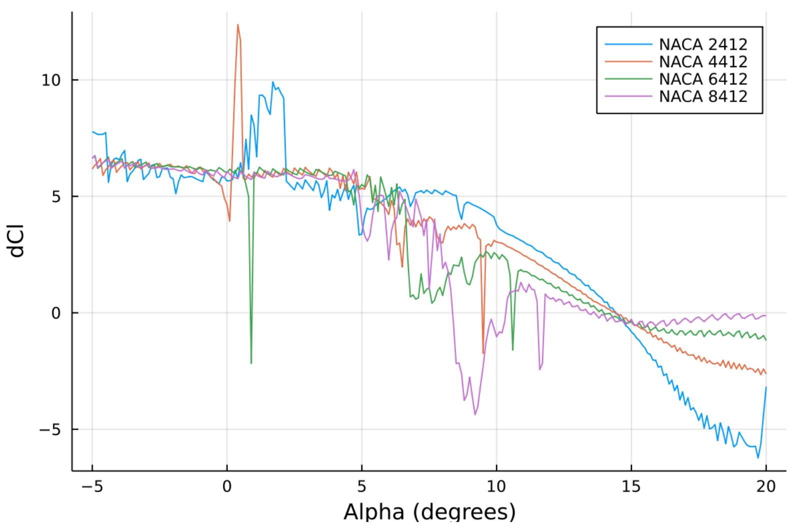

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Figure 13: Lift Derivative with Respect to Camber