Airfoil Write Up

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1 Dictionary

***Chord Line = The imaginary straight line joining the leading edge and trailing edge of an airfoil.

***Camber = This shows the asymmetry between the two acting surfaces of an airfoil (takes the average height above or below the axis with respect to the upper and lower geometry of the airfoil). A cambered airfoil is usually convex on its upper half.

***Thickness = The greatest distance between the top and lower points of the airfoil. Although the thickness of an airfoil can be found at any point along its chord line, thickness refers to the max thickness of the airfoil.

***Vortex = A region where a fluid revolves around an axis line. A vortex can induce velocities on things around it, and the magnitude of this induced velocity is inversely proportional to the distance away from the the center of the vortex.

***Freestream Velocity = This is the velocity of the fluid far enough ahead or far enough outside of the boundary layer in which the fluid hasn't been disrupted by the geometry or turbulence of the object passing through it.

***Doublet = This is the result of the construction of a flow field in which a source and a sink are placed very close to each other. The superposition of these two will result in an object being pushed away from the source or being brought in towards the sink if said object was close enough. A source provides flow going outwards from its center while a sink does the opposite and brings flow in.

***Coefficient of Drag = the ratio of the drag force on an object with respect to its velocity, the density of the fluid, and the area that the wing of an aircraft

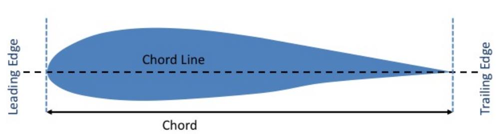

Figure 1: Chord Line

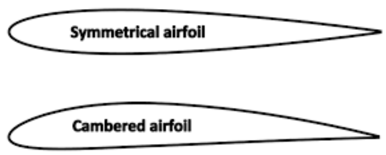

Figure 2: Cambered vs Symmetric Airfoil

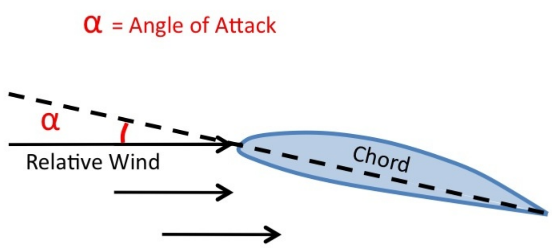

Figure 3: Angle of Attack

spans: $c_d = \frac{2F_d}{\rho A_d V_\infty^2}$

***Coefficient of Lift = the ratio of lift force on an object with respect to its velocity, the fluid density, and the area that the wing of an aircraft spans: $c_d = \frac{2F_l}{\rho A_l V_\infty^2}$

***Pitching Moment Coefficient: Gives the ratio of the moment caused by the Lift force with respect to the wing area, chord length, and the dynamic pressure: $C_m = \frac{M}{qAc}$

***Angle of Attack = This is the angle formed between an airfoil's chord line and the direction of the relative flow coming towards said airfoil. If the chord line of an airfoil is parallel to the direction of the oncoming flow, then the angle of attack would be zero: α

***Airfoil Polar = A graph or table that relates important aerodynamic coefficients, such as lift and drag, to other quantities like the angle of attack. These polars are meant to show how our desired coefficients change with respect to the angle of attack or even with respect to other coefficients (e.g. lift to drag ratio).

***Lift Curve Slope = As angle of attack changes, the lift (coefficient of lift) changes with a theoretical slope of 2π. This only occurs in the linear region, though, and when an airfoil approaches stall, this slope will approach zero, which will be followed by the lift slope becoming negative with increasing angle of attack.

***Stall = When angle of attack is increased too high past the point of maximum lift, leading to a decrease in lift. Stall comes after the linear region of the lift polar is passed: usually when the angle of attack begins to exceed 10 degrees.

***Reynold's Number = The ratio of inertial forces to viscous forces of a fluid. It describes how the fluid will react when forced to interact with the things around it. It can help show the chance that the flow around an object will separate, and it can even show how viscous a fluid will be in flow: $Re = \frac{\rho u L}{\mu}$ (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 outside the boundary layer of an object (freestream velocity) 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: With respect to a standard right hand coordinate system, imagine that the leading edge of an airfoil is at the origin and the trailing edge is somewhere on the positive x-axis. Additionally, imagine that the flow of the fluid is traveling from the negative x-axis to the positive x-axis. If the airfoil rotates with respect to the positive y-axis, that creates a positive angle of attack. On the contrary, rotation with respect to the negative y-axis will lead to a negative angle of attack. The orientation of the airfoil can change, but this sign convention will stay consistent with respect to the oncoming flow.

2 Methods for Finding Values of Xfoil.jl

The important method for finding the values that Xfoil.jl 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.jl 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.jl uses something called panel method, which, like the name implies, breaks the boundary of the airfoil into panels. Additionally, panel method has each of the point vortices and sources along the panels instead of the chord line.

3 Using Xfoil.jl

Xfoil.jl 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.jl, 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

Angle of attack is extremely import when it comes to finding the principle aerodynamic coefficients. The coefficient of lift, drag, and moment are all functions of angle of attack, and that is why aerodynamic polars usually have this value as the independent variable. 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. Furthermore, a point on the lift polar called the zero-lift angle of attack, α_0 gives us crucial information pertaining to the utility of an airfoil. This point on the polar shows at what angle of attack zero lift is produced by the airfoil. On symmetric airfoils, this angle is zero, but on cambered airfoils, this angles is less than 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.

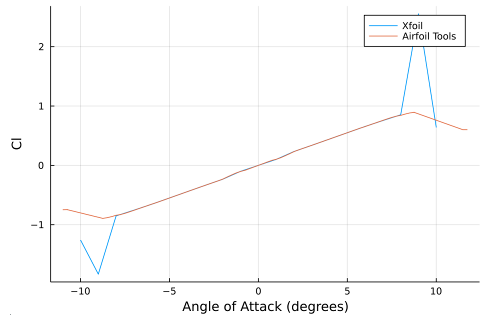

Figure 4: Coefficient of Lift Validation

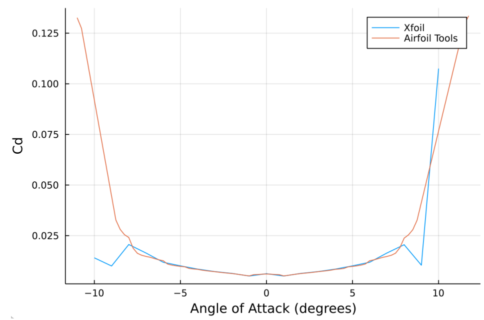

Figure 5: Coefficient of Drag Validation

Figure four shows that the data from Xfoil.jl 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.jl data are almost exactly the same.

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

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.

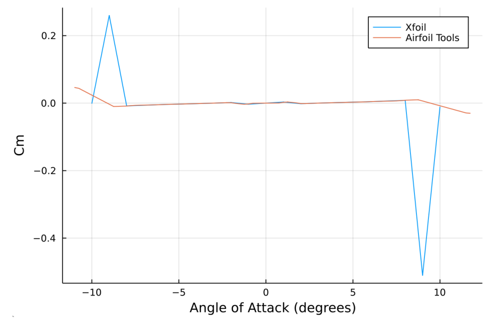

Figure 6: Coefficient of Moment Validation

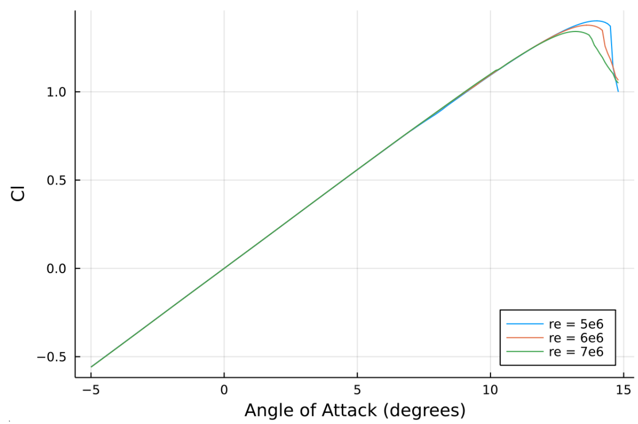

Figure 7: Reynold's Number's Effect on Coefficient of Lift

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 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.

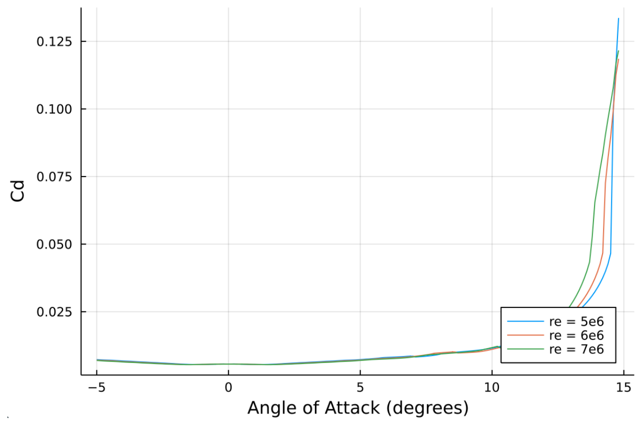

Figure 8: Reynold's Number's Effect on Coefficient of Drag

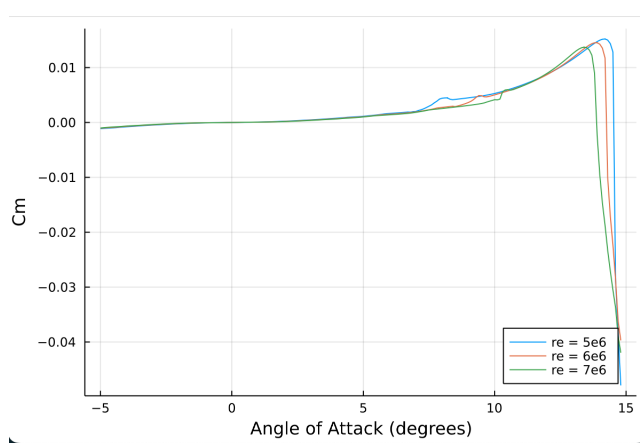

Figure 9: Reynold's Number's Effect on Coefficient of Moment

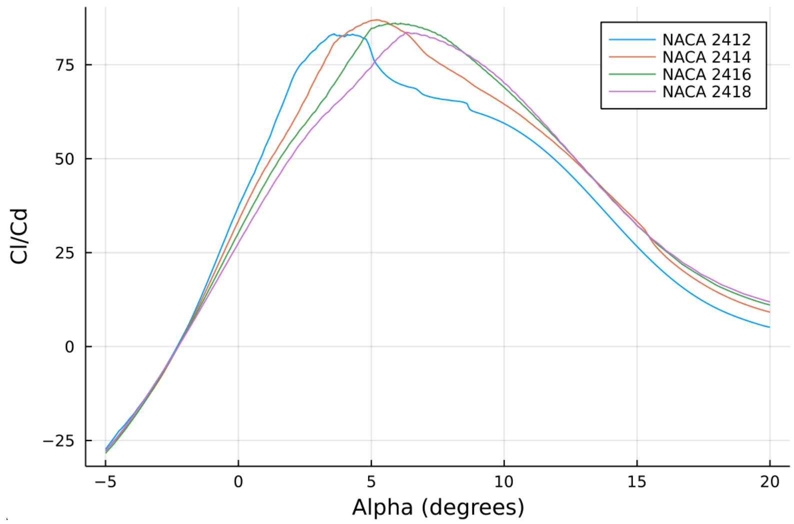

Figure 10: Lift to Drag Ratio with Respect to Thickness

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 angle of attack. This is the result of the steeper slope for the lift to drag ratio that the thinner airfoil like the NACA 2412 has after the zero lift angle. Furthermore, thicker airfoils can hold a higher lift to drag ratio following their maximum while the thinner airfoils decrease at a much faster rate with respect to angle of attack.

The important thing that figure eleven shows is that in the linear region, the 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. This higher lift slope before stall was seen in the lift to drag polar in figure 10. It was determined that the thinner airfoils were able to have their lift to drag ratios increase at a faster rate with respect to angle of attack before stall. This makes sense physically since a thicker airfoil will have a larger area for drag to act upon while the thinner airfoils can stay more stream line: even at higher angles. However, the thinner airfoils had noticeably faster drop offs in the lift to drag ratio following stall. This was most likely the result of fluid separation towards the trailing edge. A thinner airfoil would require the flowing fluid to make sharper turns with respect to the airfoil geometry as is passes around it. This would make it much easier for the fluid to separate, leading to significant losses in lift. A larger airfoil has a more obtuse shape, making it harder for the fluid to separate as it passes by.

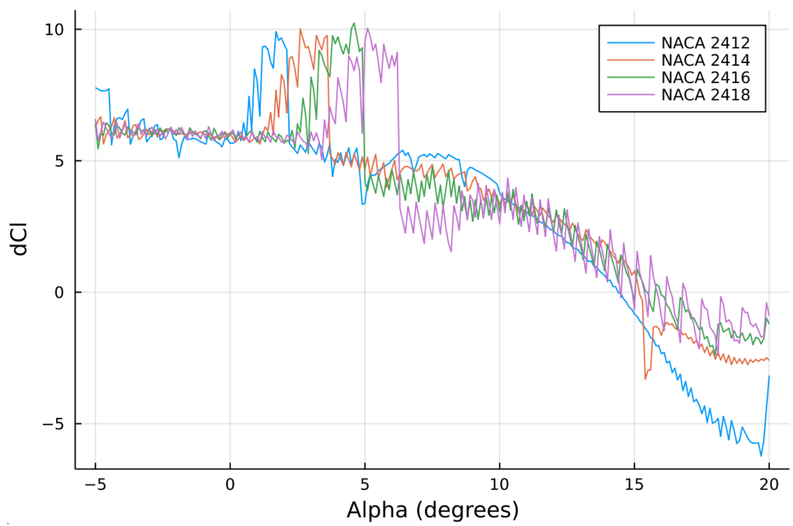

Figure 11: Lift Derivative with Respect to Thickness

The main takeaway from figure twelve is that a lower cambered airfoil has a lower maximum lift to drag ratio. The reasoning for this has everything to do with the geometry of a more cambered airfoil. The convex design of the upper edge creates a more favorable environment for faster moving air on the top surface and slower moving air on the bottom. As Bernoulli's equation shows, faster moving air produces lower pressures while slower moving air is the opposite. This allows for the more cambered airfoil to have a higher pressure pushing underneath it while the fast moving air on top doesn't react with as high of magnitude. This leads to the great lift generation from a cambered airfoil. Furthermore, at larger angle of attacks, a lower cambered airfoil has a higher lift to drag ratio than the higher cambered ones. Additionally, figure twelve also shows that as camber increases, the zero lift angle decreases significantly.

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.

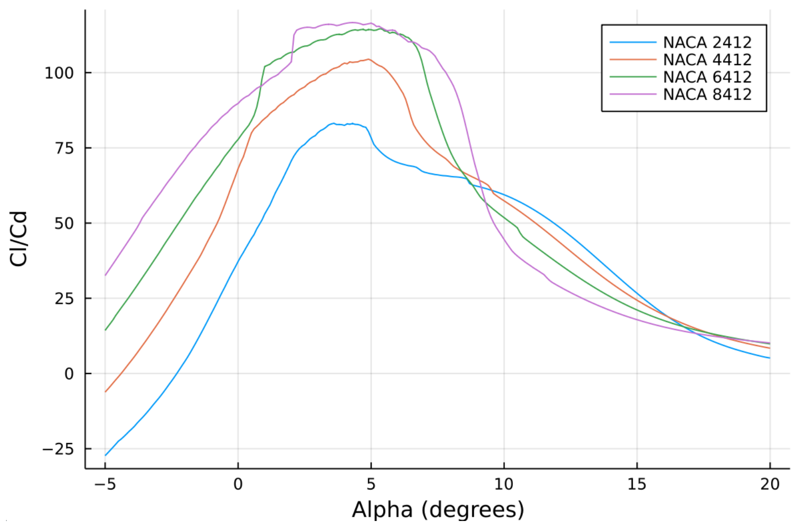

Figure 12: Lift to Drag Ratio with Respect to Camber

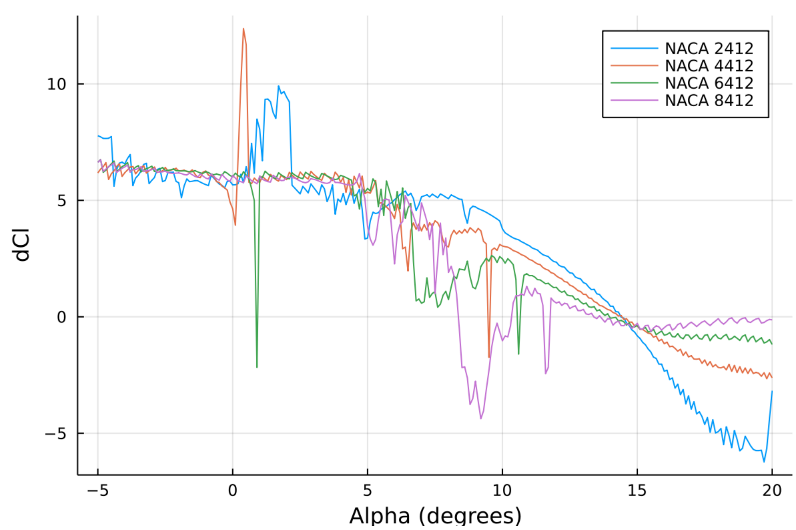

Figure 13: Lift Derivative with Respect to Camber