ME 163 | Engineering Aerodynamics | Prom Putthisri
Spring 2021 | Prof. Omer Savas | Project 3

1 Abstract

Currently, the conventional method of designing and studying airfoil is to refer back to collected experimental result. The objective of this project is provide an alternative to the study of airfoil through computational tools as an estimation of empirical data sets. Utilizing vortex panel method, Kutta's condition, lift/drag/-pressure coefficient relationship, and Thwaites method, our goal is to build and compare our computational fluid dynamic model for NACA-0012 and NACA-4412 airfoil at angle of attack -16° ¡ α ¡ 16°.

2 Project Materials

2.1 Pressure Coefficient Distribution

Coefficient of Pressure, C_p, is a crucial parameter for both Thwaites and Stratford methods. In project, the flow used is a uniform flow with constant velocity of 1 m/s. The airfoil is created through the method from section 9.5 of the lecture note with 256 control points and chord length of 1 m. Using vortex panel method from project 2, we are able to acquire C_p.

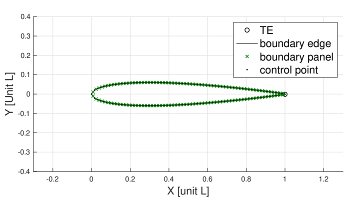

Figure 1: Self-Generated 0012-NACA

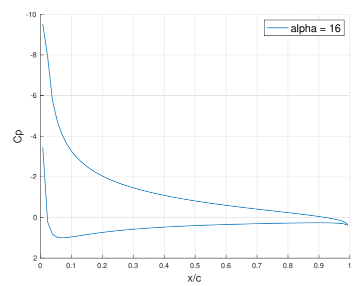

Figure 2: 0012-NACA C_p upper and lower surfaces at $\alpha = 16^\circ$

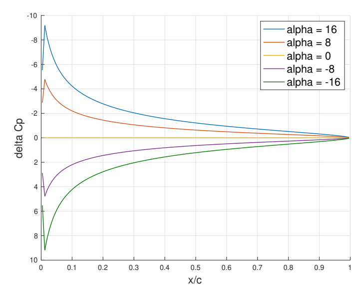

Figure 3: 0012-NACA ΔC_p at Five α

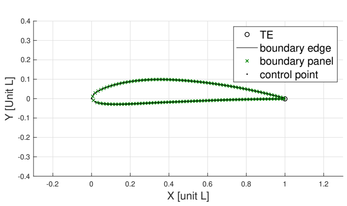

Figure 4: Self-Generated 4412-NACA

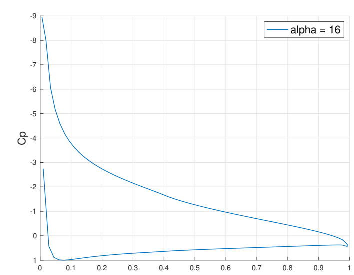

Figure 5: 4412-NACA C_p of upper and lower surfaces at $\alpha = 16^\circ$

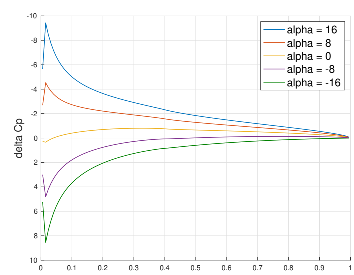

Figure 6: 4412-NACA ΔC_p at Five Different α

2.2 Thin Airfoil and Cambered Airfoil Theory

The flow at the surfaces of a body may be modeled as a vortex sheet wrapping around it. Each vortex, control point, will experience the effect of the flow differently. Thus, having different forces and coefficient related to it. Fluid is chaotic by nature, but surprisingly we are able to predict its behavior through numerical and analytical methods. From thin airfoil theory, the numerical Lift is equal to $\rho U\Gamma$ or $\pi\alpha\rho cU^2$. From cambered airfoil theory, in section 9.8 of the course note, numerical Lift can be calculated from Area sweeping of airfoil and the dynamic pressure following the formula below.

$$L_{numerical} = \rho U\Gamma = \pi\rho cU^2(A_0 + \frac{1}{2}A_1)$$

The numerical lift coefficient is defined as Lift divided by the dynamic pressure and chord length, thus we have the following.

$$C_l = \frac{\rho U\Gamma}{\frac{1}{2}\rho U^2 c} = 2\pi(A_0 + \frac{1}{2}A_1)$$

To obtain formula for analytical value, we are required to derived $C_{Lnumerical}$ in terms of C_p that we have acquired from the vortex panel method. Utilizing forces and method from coding assignment 3, we have the following.

$$[F] = \int_A (\Delta P + P_\infty)\cdot\hat{n}\cdot dA$$
$$C_p = \frac{(\Delta P + P_\infty)}{\frac{1}{2}\rho U_\infty^2}$$

Dividing the dynamic pressure and Chord Length, c, across the formula for forces we will get coefficient of force along the normal direction in terms of C_p. We then changes the domain of area to panel area and break the coefficient of force into its Cartesian's component.

$$\frac{[F]}{\frac{1}{2}\rho U_\infty^2} = \int_A C_p\cdot\hat{n}\cdot\frac{dA}{c}$$
$$[C_{Fx}, C_{Fy}] = \int_S C_p(cos(\theta_s + \frac{\pi}{2}), sin(\theta_s + \frac{\pi}{2}))\cdot\frac{dS}{c}$$

By definition, the flow path and its orthogonal is the direction of drag and lift accordingly. Therefore, rotating the components of forces along the angle of attack would result in the numerical coefficient of lift and drag in term of C_p.

$$[C_d, C_l] = [C_{Fx}, C_{Fy}]\cdot\begin{bmatrix}\cos(\alpha) & \text{-}\sin(\alpha)\\ \sin(\alpha) & \cos(\alpha)\end{bmatrix}$$

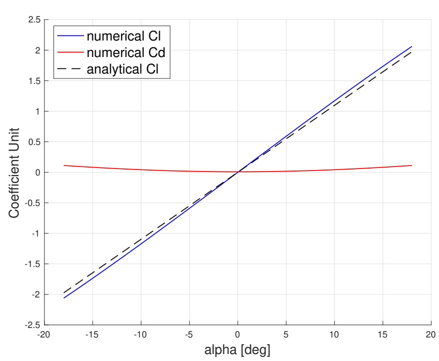

Figure 7: 0012-NACA numerical vs. analytical C_l

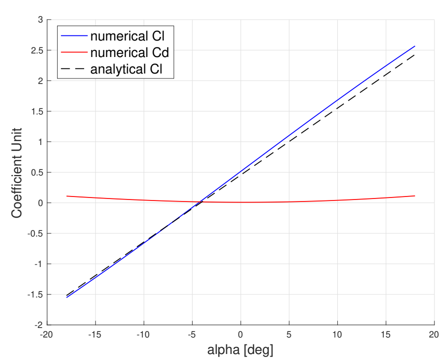

Figure 8: 4412-NACA numerical vs. analytical C_l

NOTE: From the discussion above, we can scale these coefficient into forces by multiplying the dynamic pressure and the chord length that we have used in the denominator above.

2.3 Separation Points and Stall

Using Thwaites' method from project 2, we are able to find separation points on the upper surface of the airfoil, which scatter through out the surface. The figure below show region where separation points occur varying from angle of attack. As angle of attack increases the closer the separation points toward the leading edge. More detail will be shown in the next sets of figures. The color gradient represent the angle of attack. The brighter the green the higher the angle of attack is.

In order to deduce the location of the stall angle from the separation point, we will need to find the angle of attack in which the separation points passes threshold distance of 0.2c of the chord length. This method is an empirical estimation and is by no means the standard method of finding the stall angle. The threshold distance is represented in the horizontal red line.
The stall angle of 0012-NACA is approximately 5.45 degrees.
The stall angle of 4412-NACA is approximately 9.15 degrees.

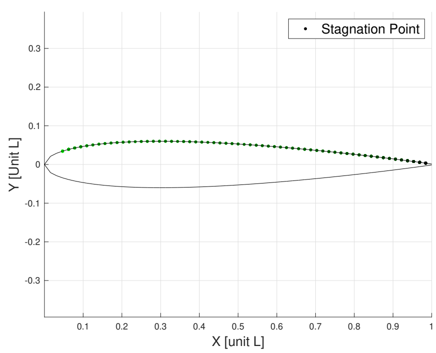

Figure 9: 0012-NACA Separation Points

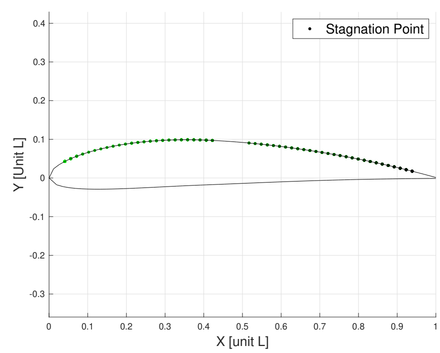

Figure 10: 4412-NACA Separation Points

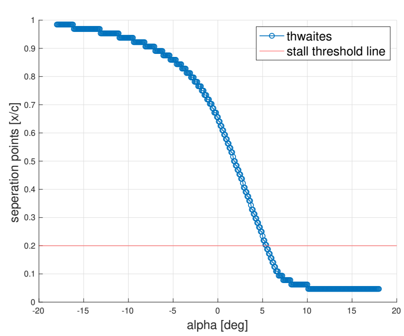

Figure 11: 0012-NACA Stall Determination Graph

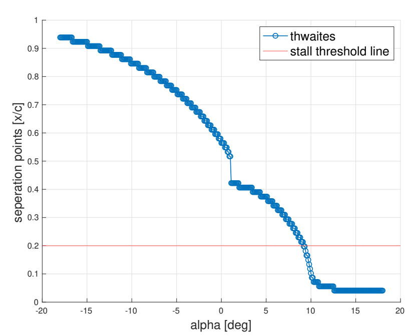

Figure 12: 4412-NACA Stall Determination Graph

2.4 Stall and How It's Actually Done

While our DIY-Computational Fluid Dynamic model gives reasonable result, they are definitely not practical. The standard designing method is still look up values from the experimental data sets. In the case of stall angle, we can find the empirical value by tracing back to the empirical C_l against α graph and determine the peak of the graph. In most case, there will be multiple data sets published. Therefore, one ca argue for many stall angle depending on the data sets.
NOTE: Example of empirical data sets is on the next page.

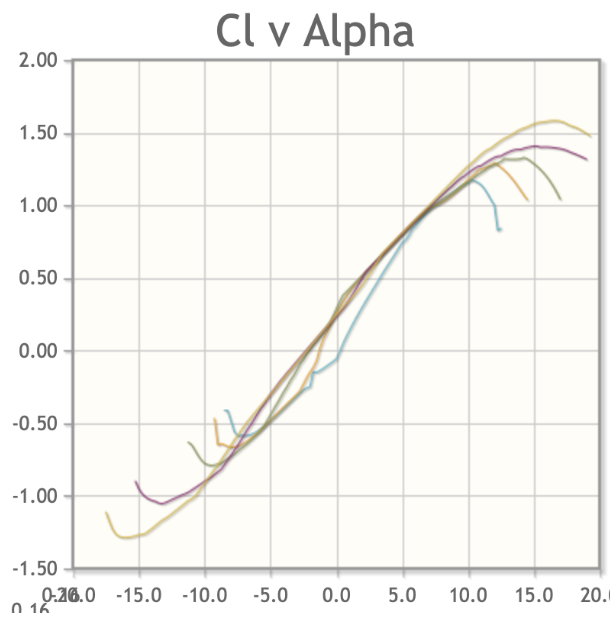

Figure 13: 0012-NACA numerical vs. analytical C_l