Lecture # 23: Airfoil Aerodynamics – Part 0 1: Airfoil Nomenclature

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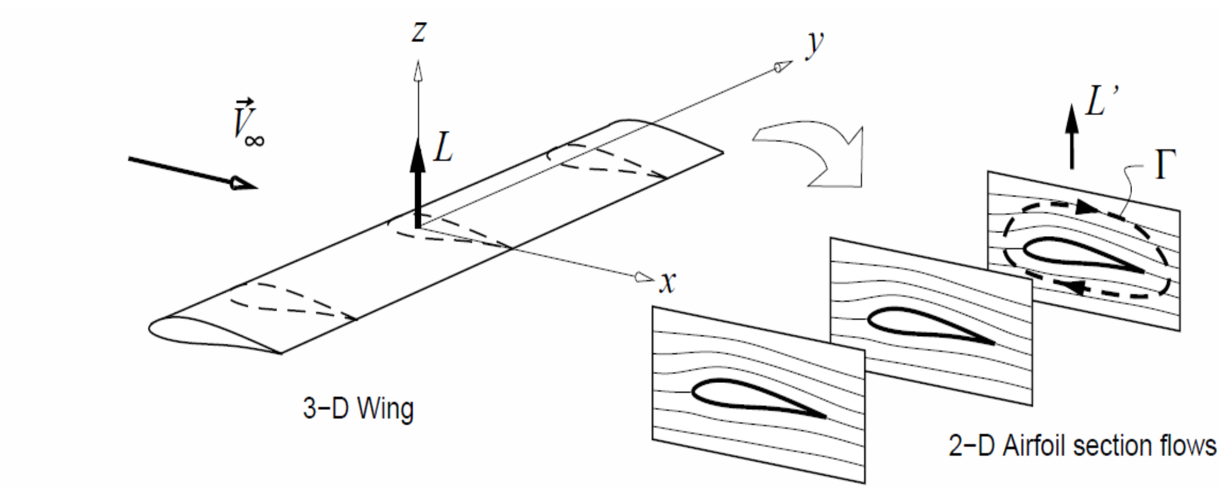
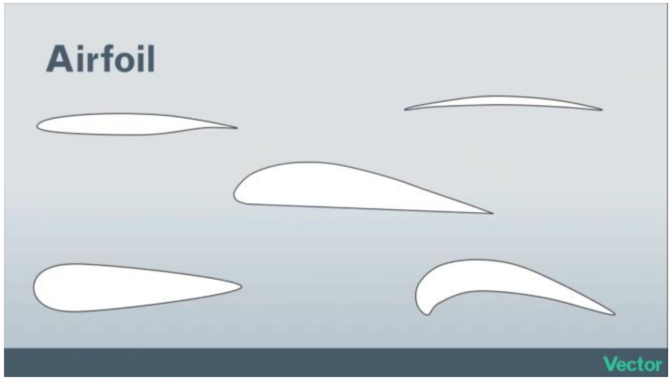
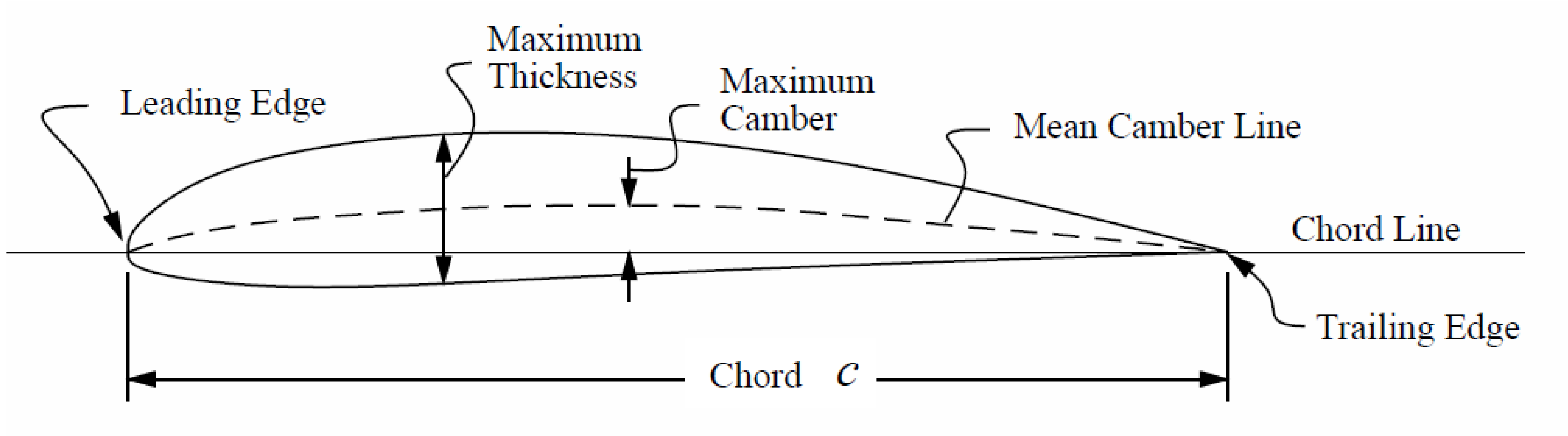
❑ Airfoil Parameter Nomenclature



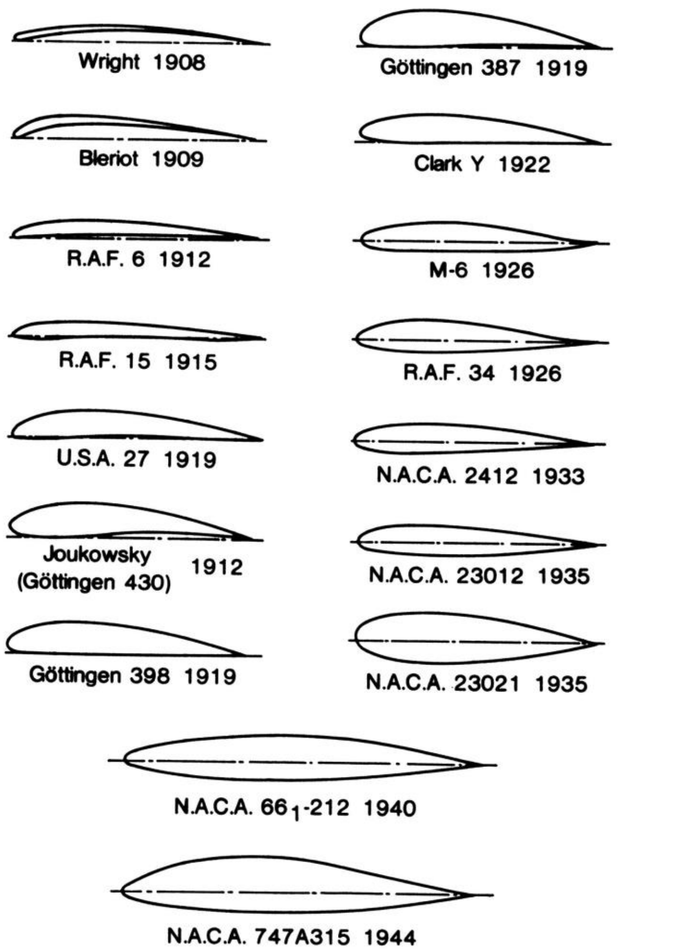




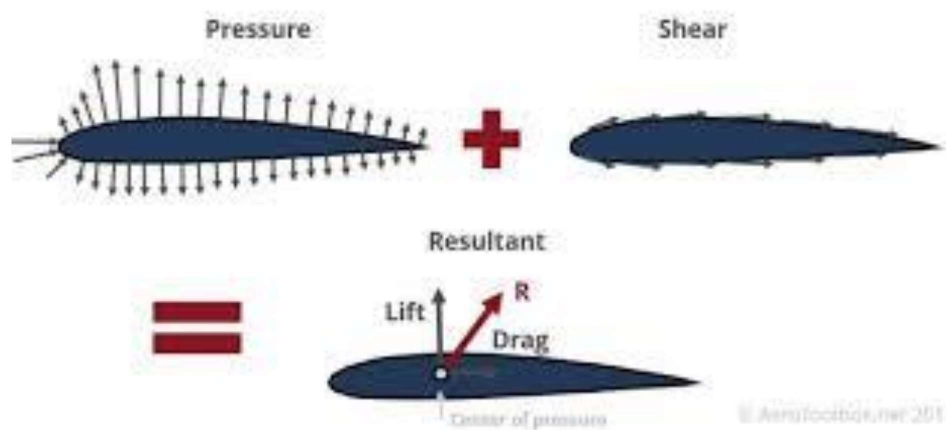
- *The **Mean Camber Line** is defined to lie halfway between the upper and lower surfaces.*

❑ Airfoil Aerodynamics

With a View to Practical Solutions



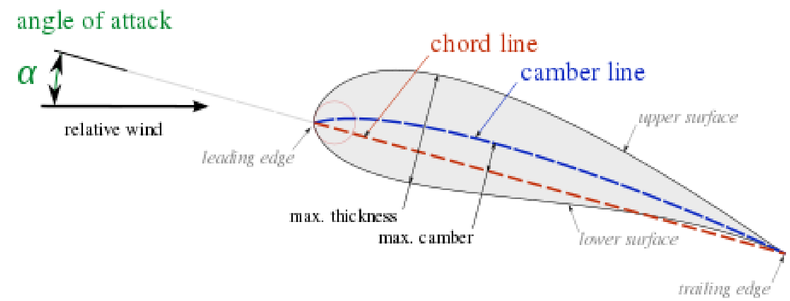
The historical evolution of airfoil sections, 1908–1944. The last two shapes (N.A.C.A. 66_1-212 and N.A.C.A. 747A315) are low-drag sections designed to have laminar flow over 60 to 70 percent of chord on both the upper and the lower surface. Note that the laminar flow sections are thickest near the center of their chords.



❑ AIRFOIL AERODYNAMICS

NACA four-digit airfoil sections define the profile by:

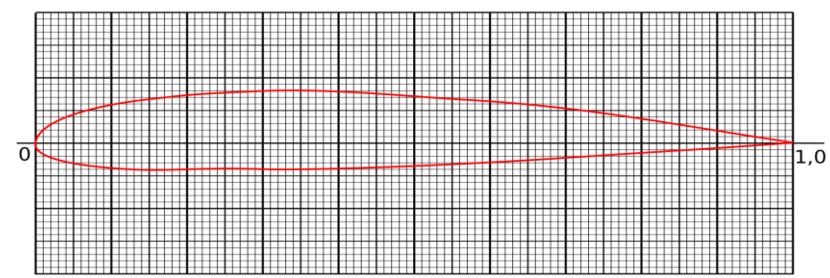
- *First digit describing maximum [camber]() as percentage of the [chord]().*
- *Second digit describing the distance of maximum camber from the airfoil leading edge in tenths of the chord.*
- *Last two digits describing maximum thickness of the airfoil as percent of the chord.*[3]



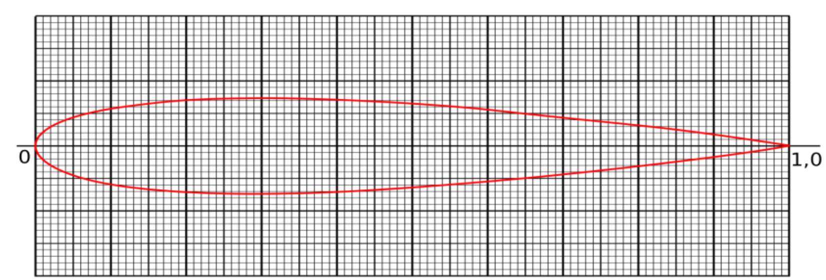
For example:

- *NACA 2412 airfoil has a maximum camber of 2% located 40% (0.4 chords) from the leading edge with a maximum thickness of 12% of the chord.*
- *NACA 0015 airfoil is symmetrical, the 00 indicating that it has no camber. The 15 indicates that the airfoil has a 15% thickness to chord length ratio: it is 15% as thick as it is long.*

NACA 2412

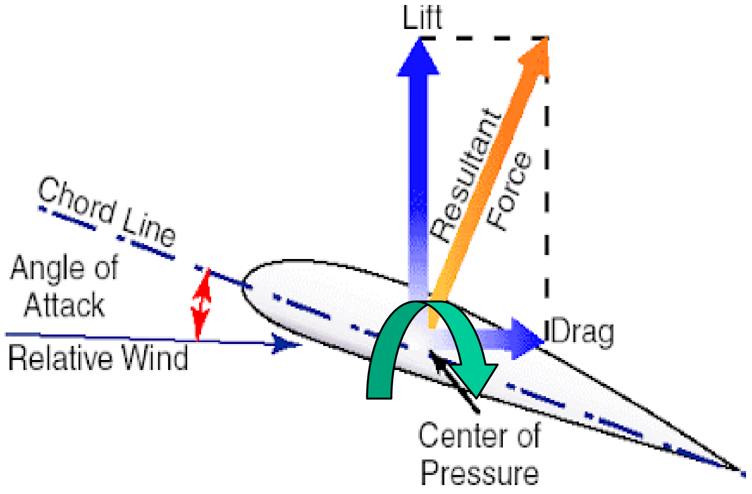


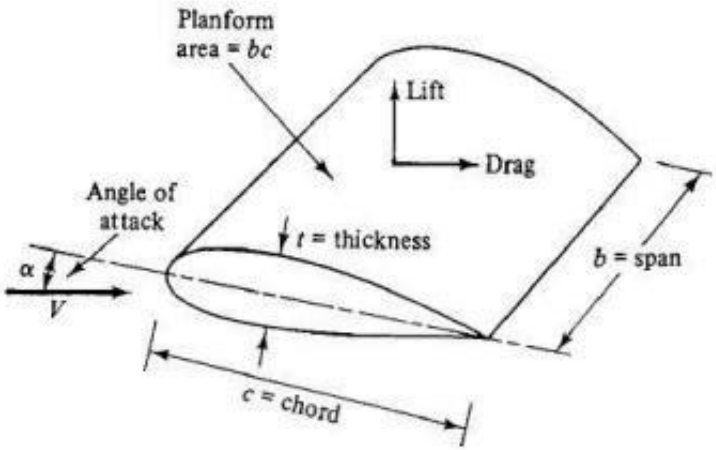
NACA 0015



Further information about NACA airfoil can be found at: https://en.wikipedia.org/wiki/NACA_airfoil

❑ Aerodynamic Performance of an airfoil





The forces and moment are more conveniently nondimensionalized using the freestream dynamic pressure $q_\infty \equiv \frac{1}{2}\rho_\infty V_\infty^2$ and the chord c, giving the lift, drag, and moment coefficients.

$$c_\ell \equiv \frac{L'}{q_\infty\, c} \quad , \qquad c_d \equiv \frac{D'}{q_\infty\, c} \quad , \qquad c_m \equiv \frac{M'}{q_\infty\, c^2}$$

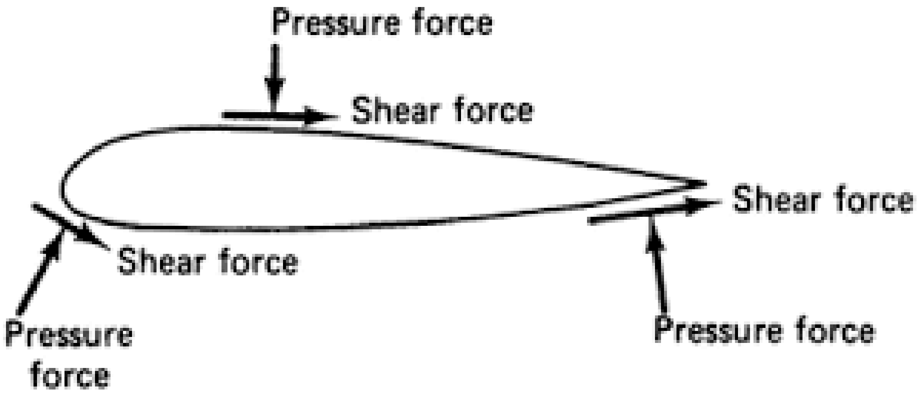
Dimensional analysis reveals that these will depend only on the angle of attack α, the Reynolds number $Re \equiv \rho_\infty V_\infty c/\mu_\infty$, the Mach number $M_\infty \equiv V_\infty/a_\infty$, and on the airfoil shape.

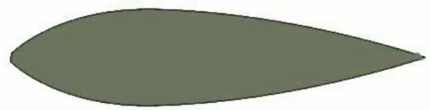
$$c_\ell\,,\, c_d\,,\, c_m \;=\; f\,(\,\alpha\,,\, Re\,,\, M_\infty\,,\, \text{airfoil shape}\,)$$

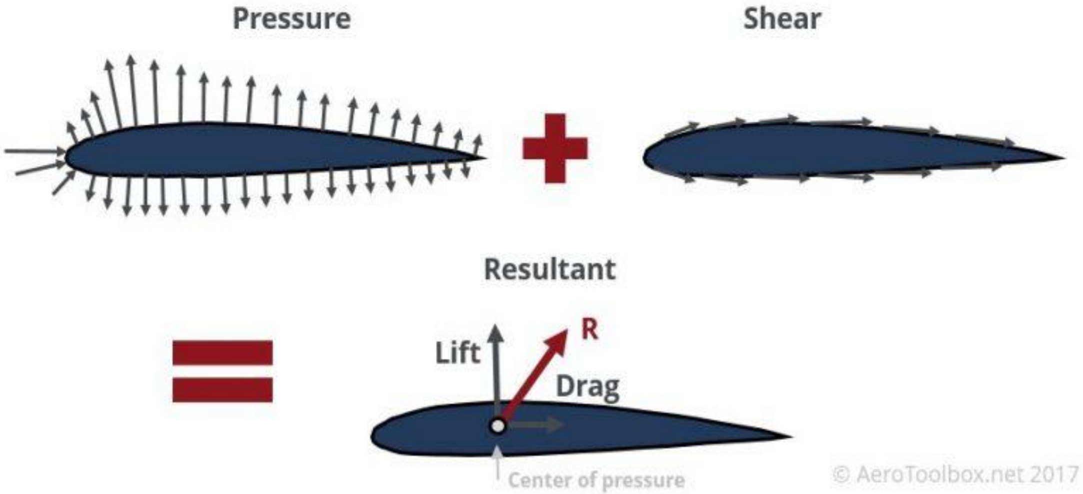
For low speed flows, M_∞ has virtually no effect. And for a given airfoil shape, we therefore have

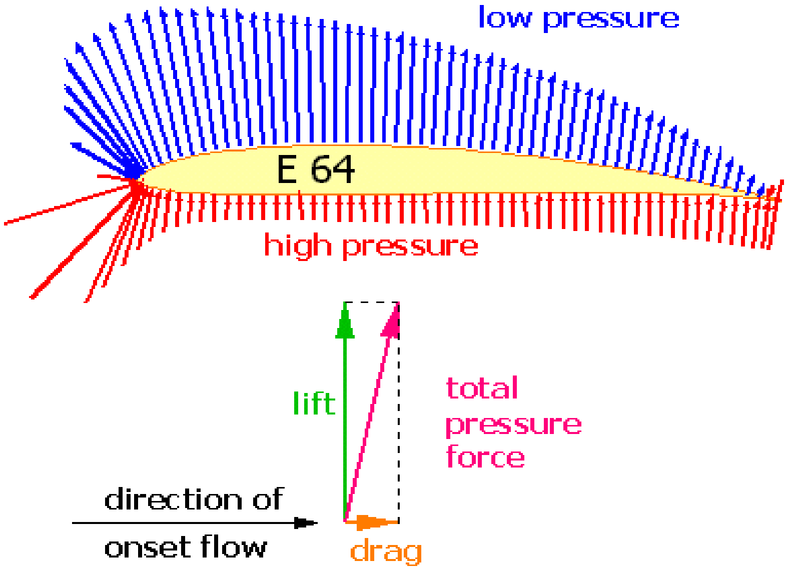
$$c_\ell\,,\, c_d\,,\, c_m \;=\; f\,(\,\alpha\,,\, Re\,) \qquad \text{(low speed flow, given airfoil)}$$

❑ Airfoil Aerodynamics

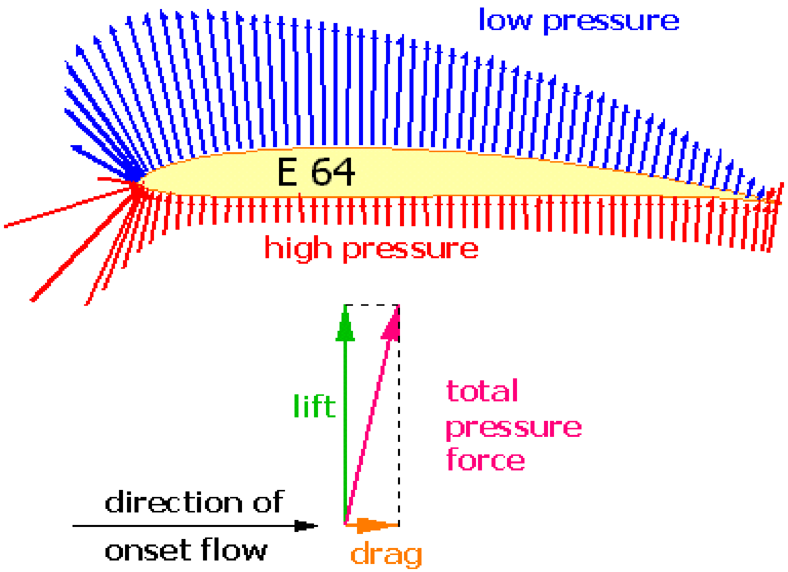


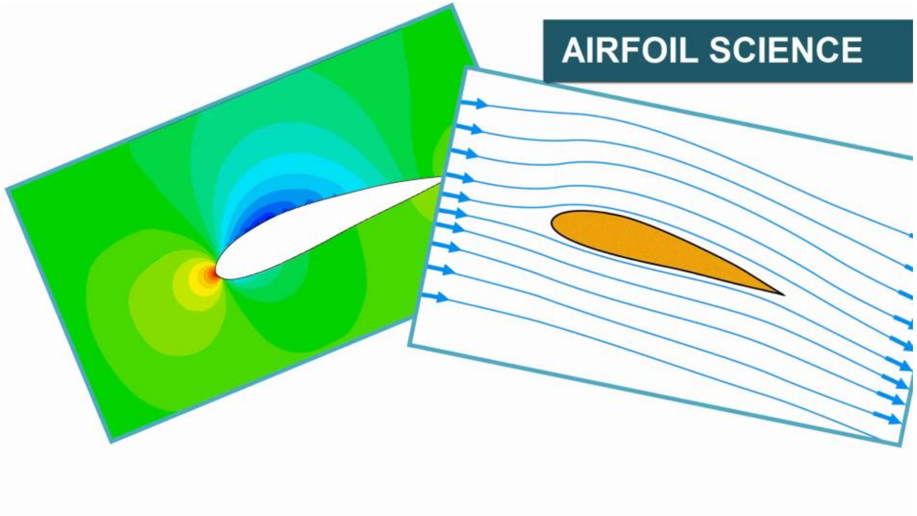


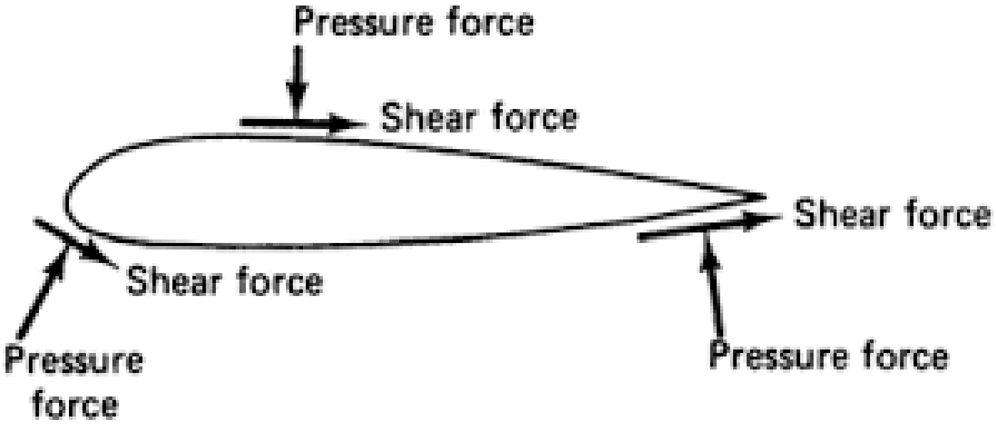




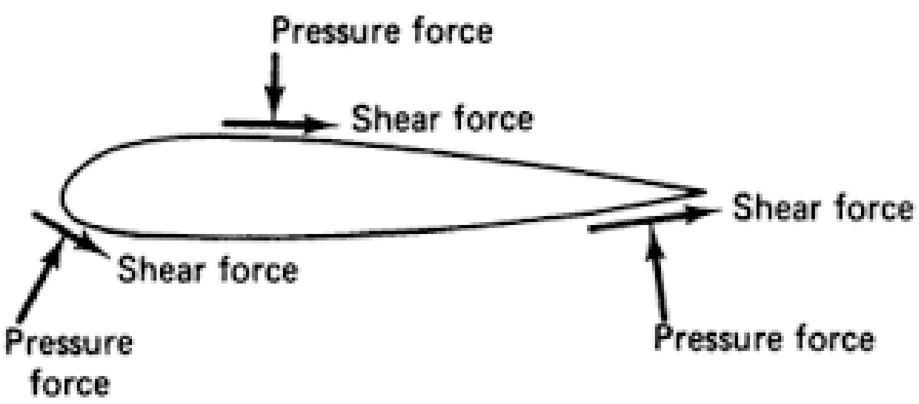
❑ Airfoil Aerodynamics

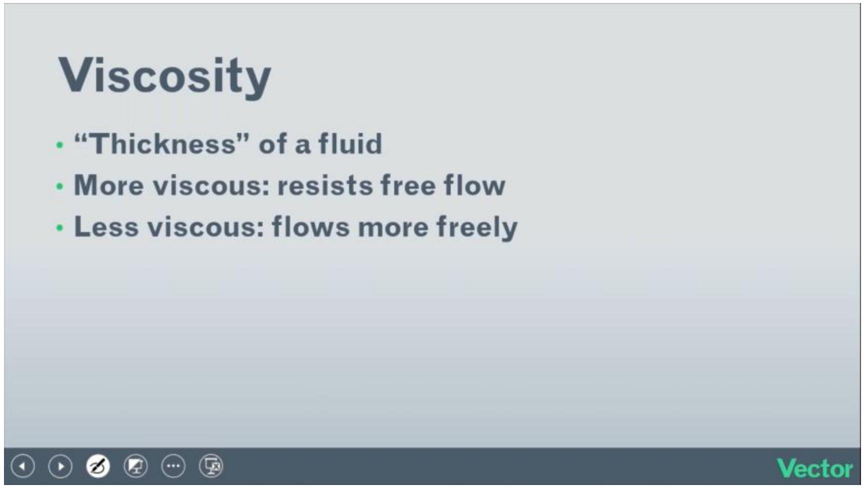


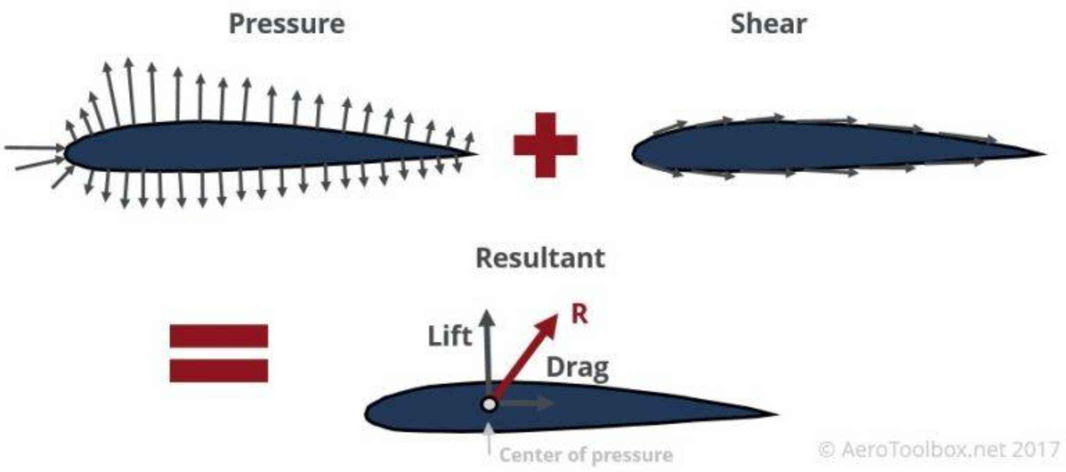


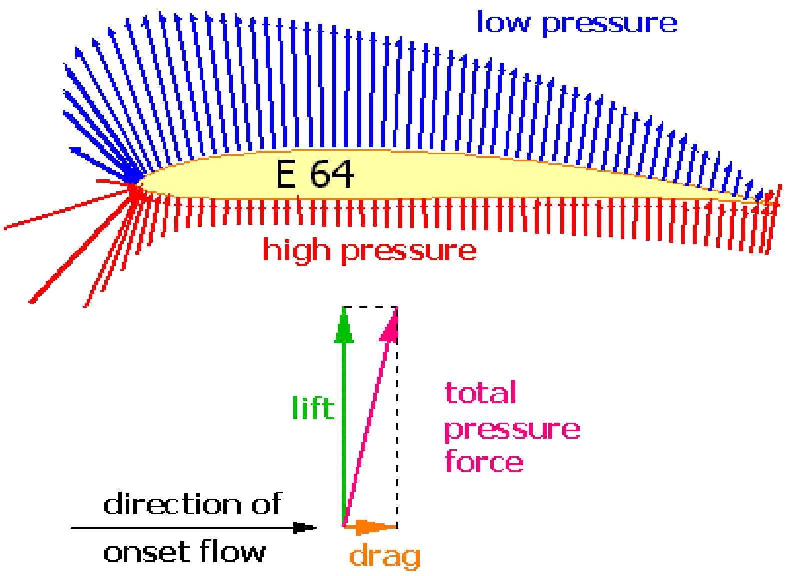


❑ Airfoil Aerodynamics



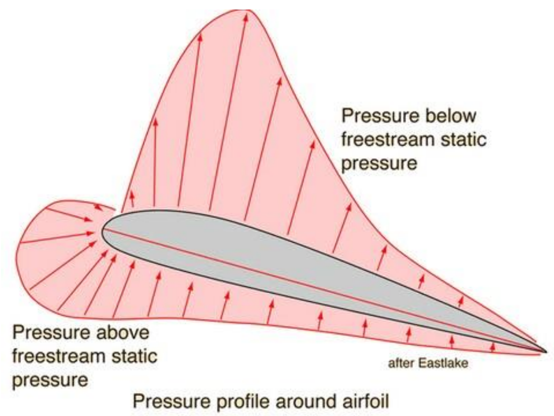


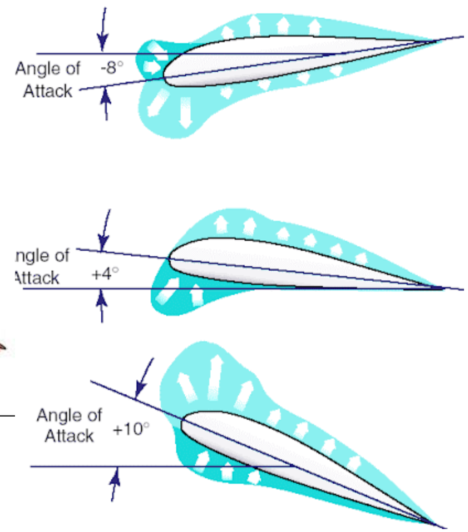


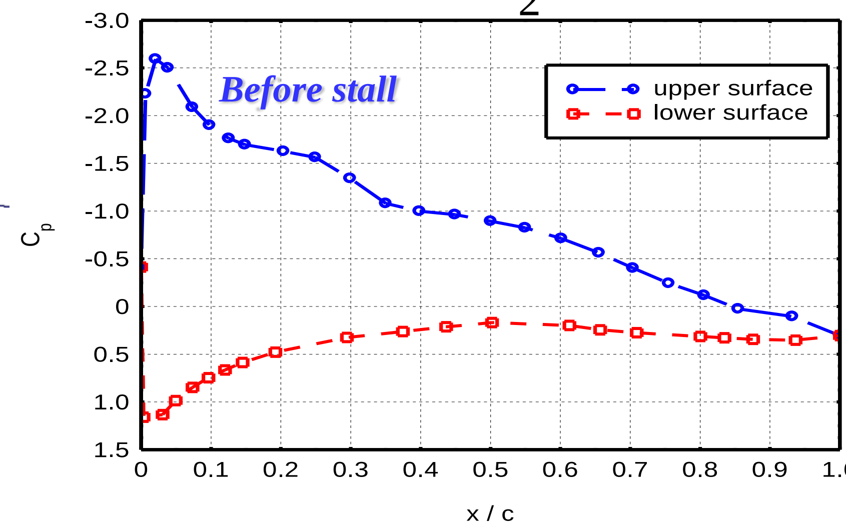


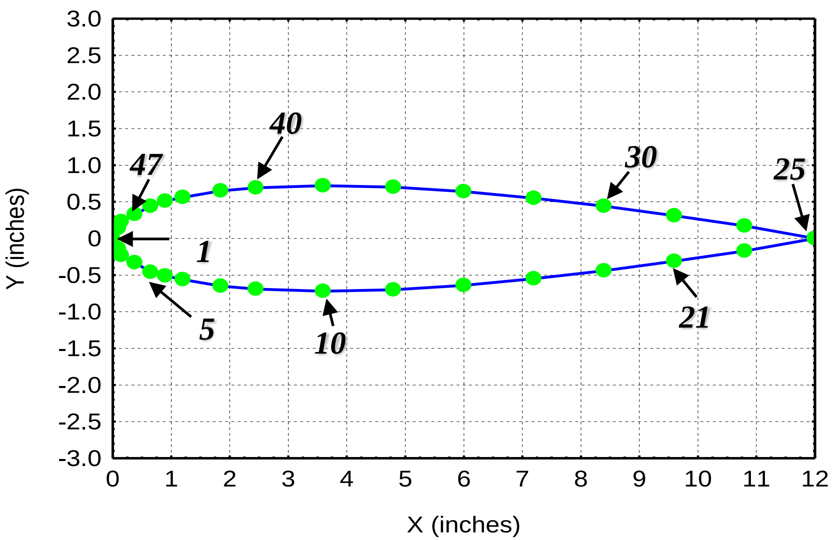
Pressure Distributions over an Airfoil

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho V_\infty^{\ 2}}$$

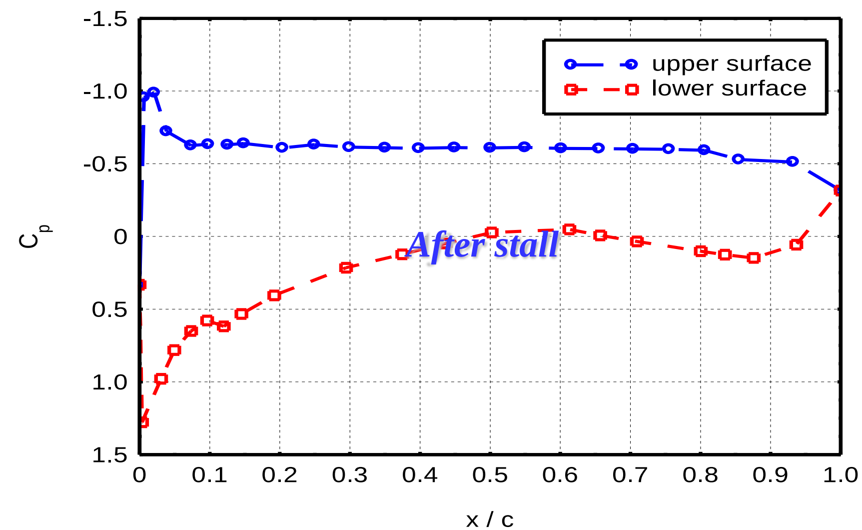




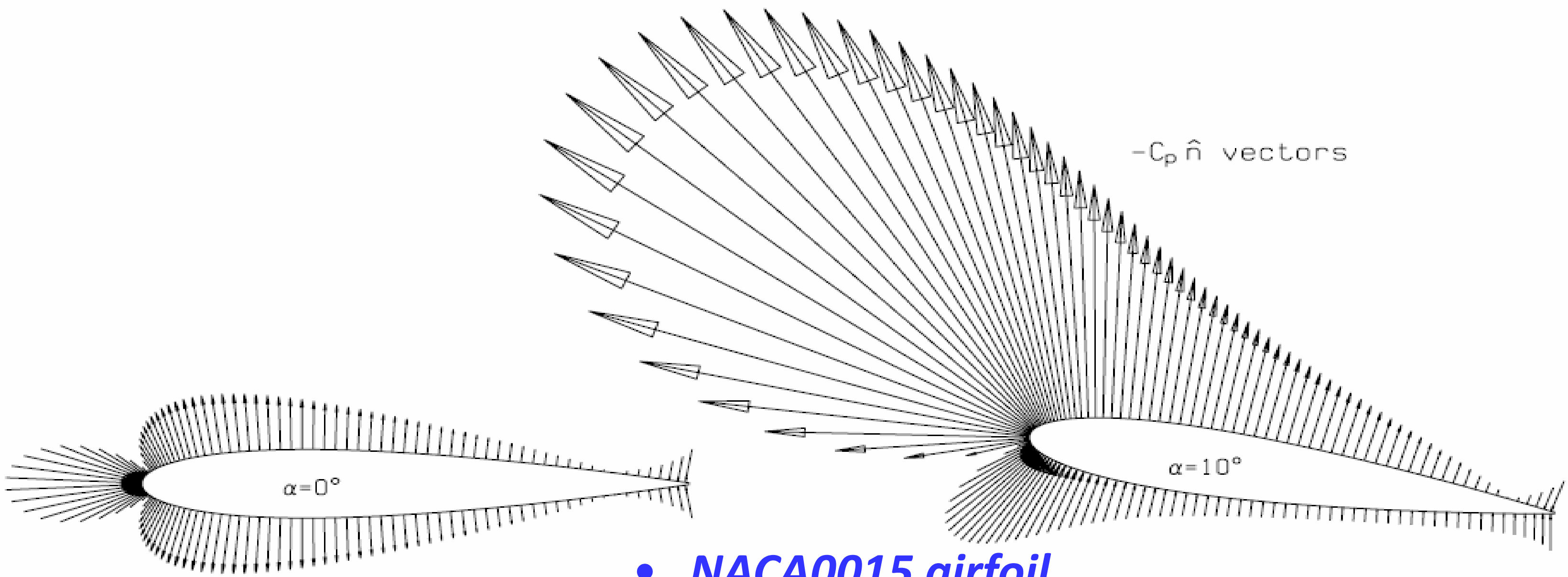




NACA0012 airfoil with 49 pressure tabs

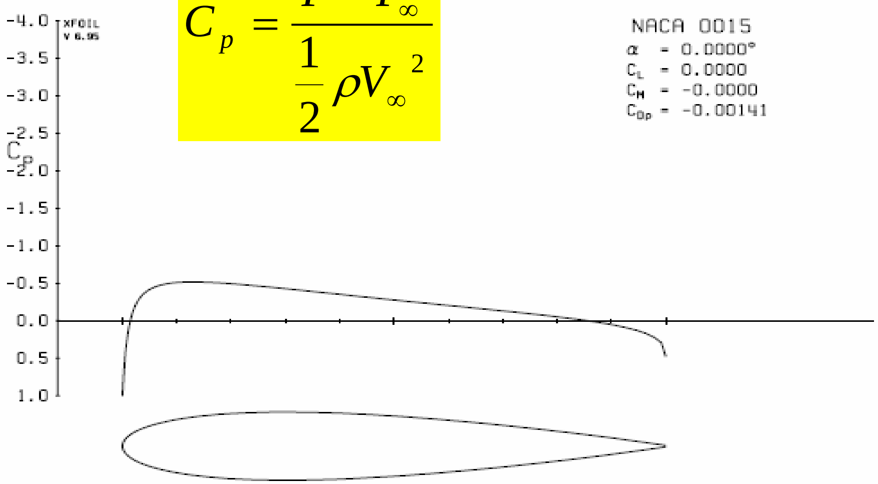


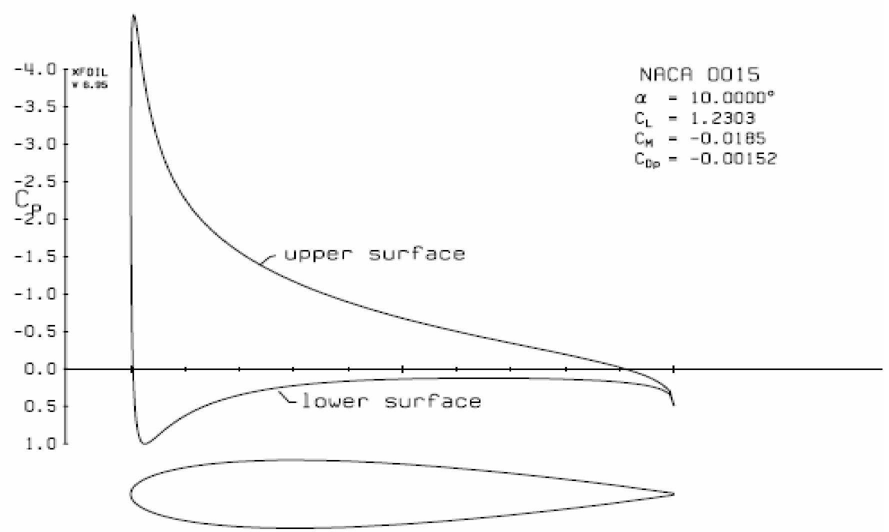
❑ Pressure Distributions over an Airfoil



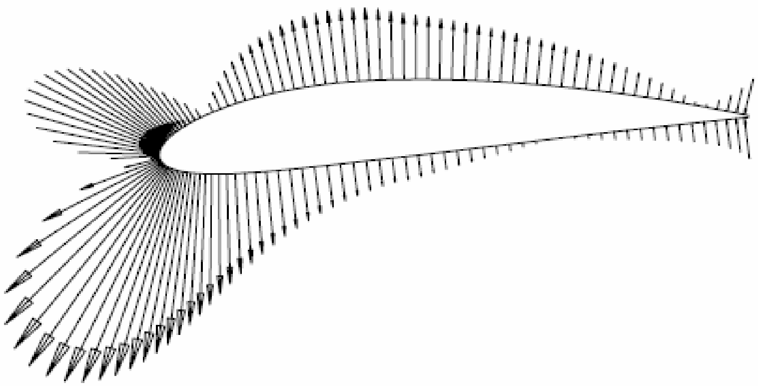
- ***NACA0015 airfoil***

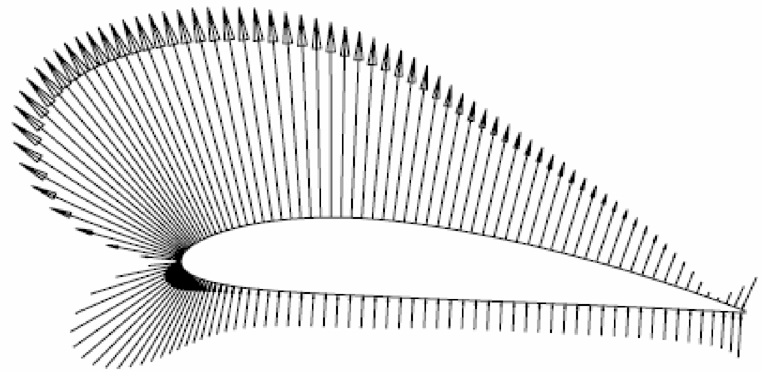
$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho V_\infty^{\ 2}}$$





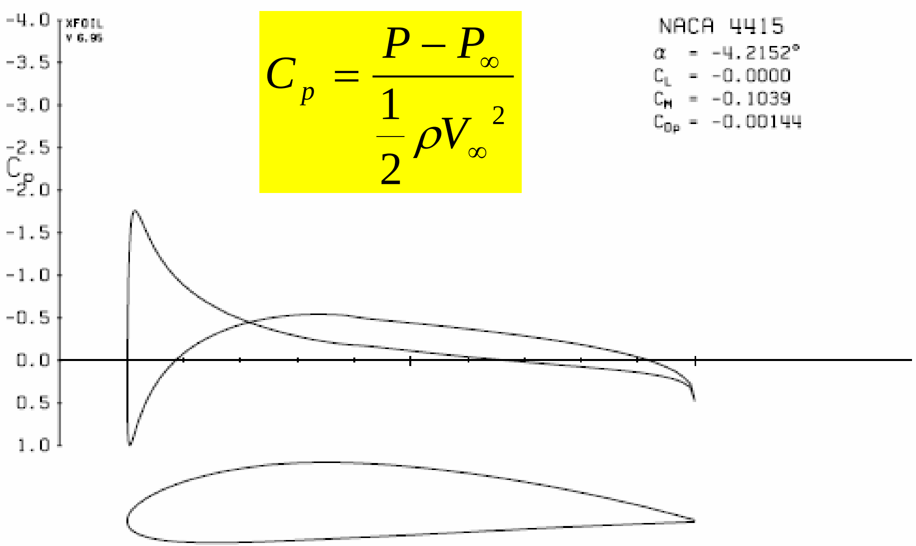
❑ Pressure Distributions over an Airfoil

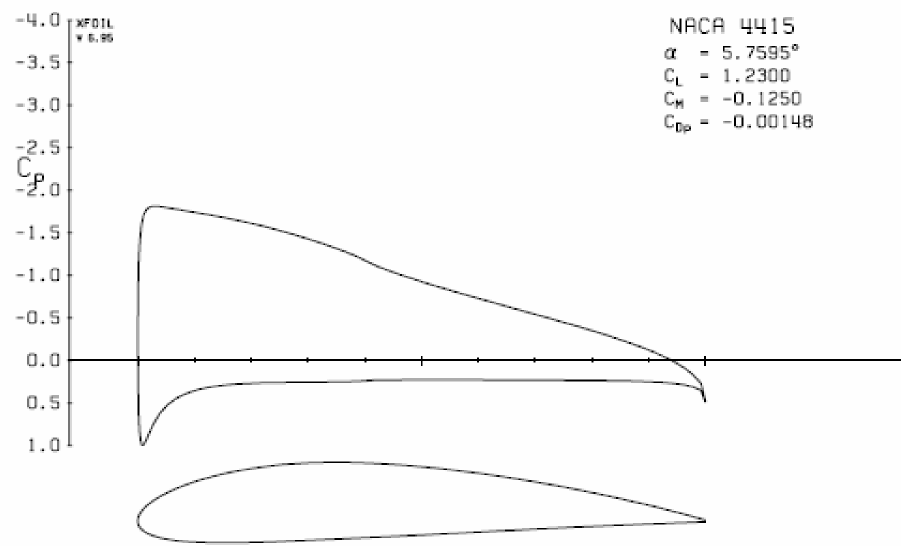




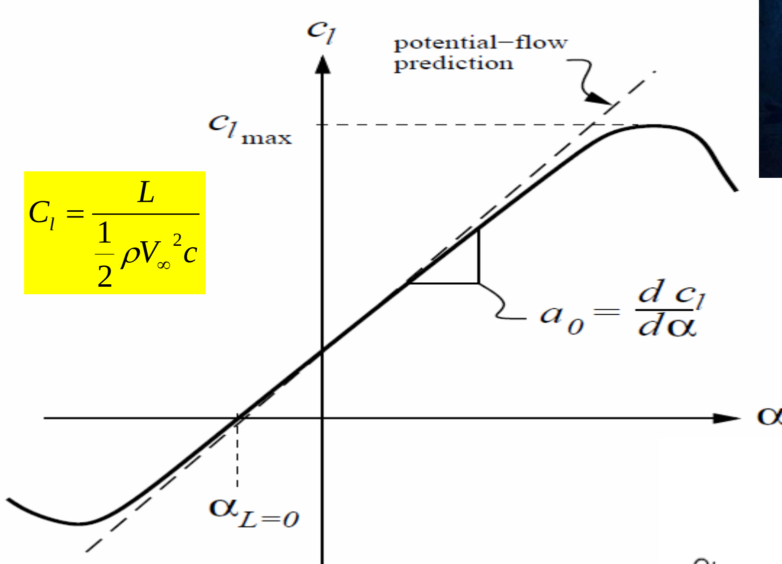
- *NACA4415 airfoil*

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho V_\infty^{\ 2}}$$

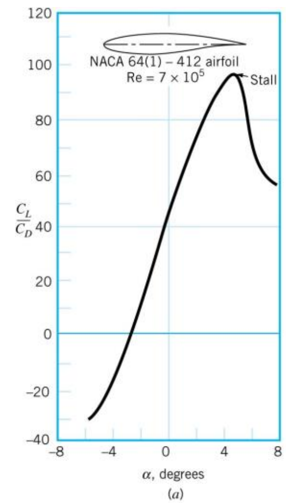


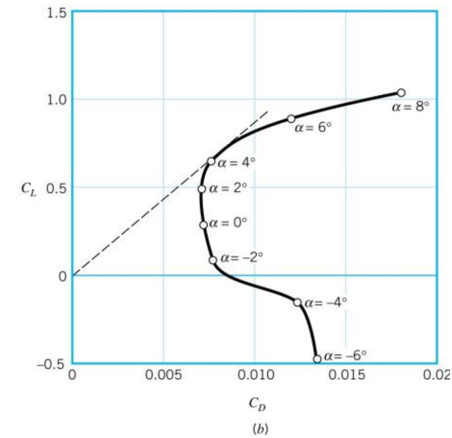


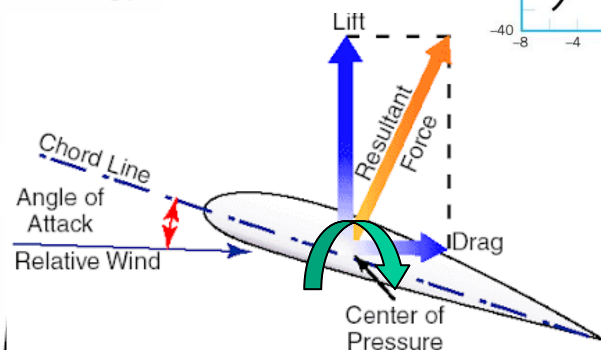
❑ Aerodynamic Performance of an airfoil

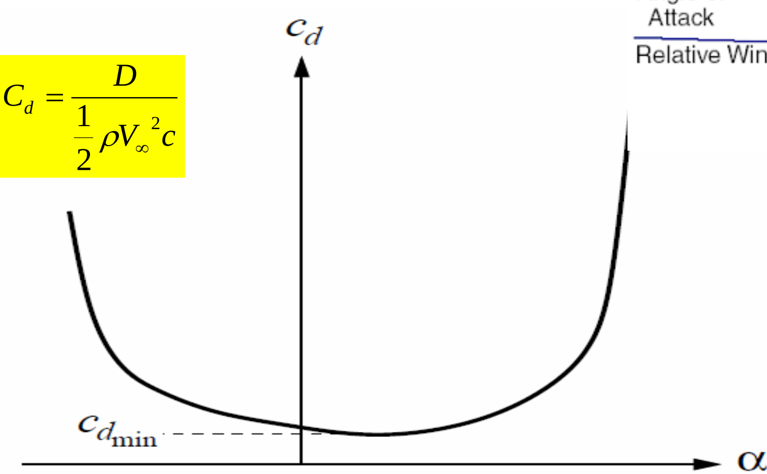


$$C_l = \frac{L}{\frac{1}{2}\rho V_\infty^2 c}$$

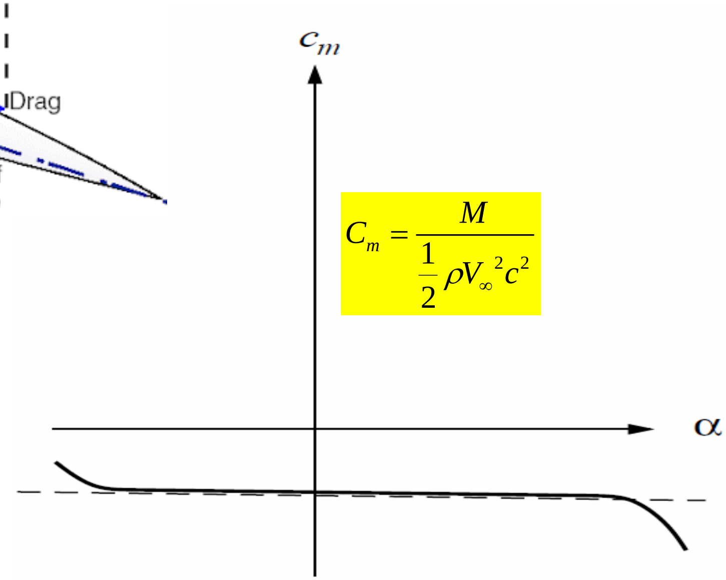






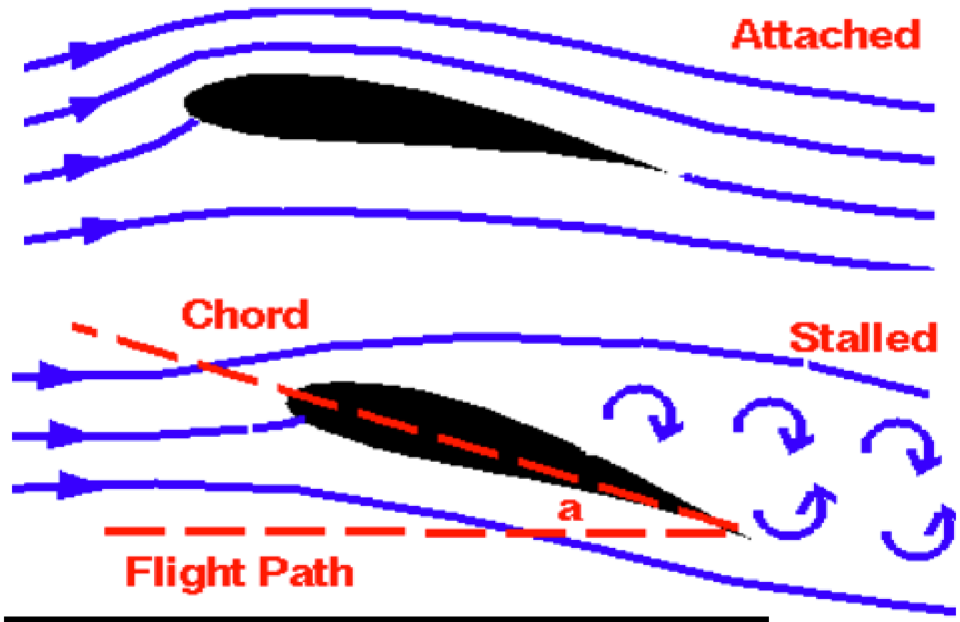


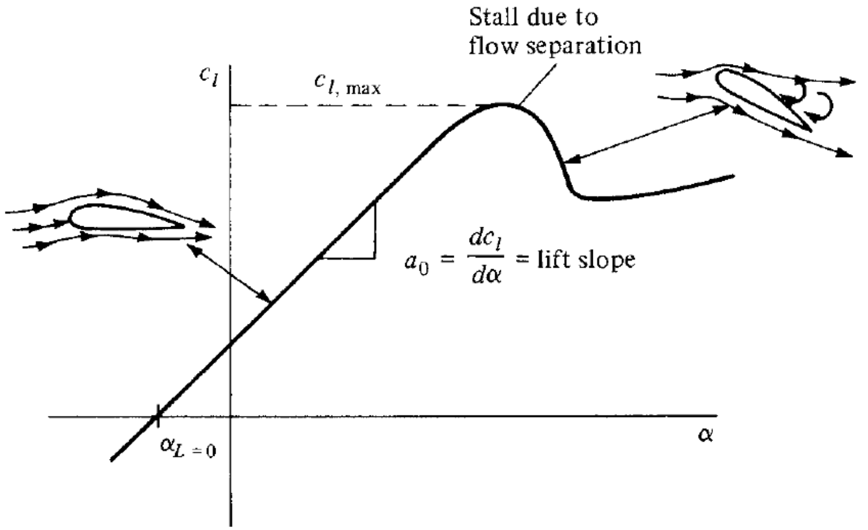
$$C_d = \frac{D}{\frac{1}{2}\rho V_\infty^2 c}$$

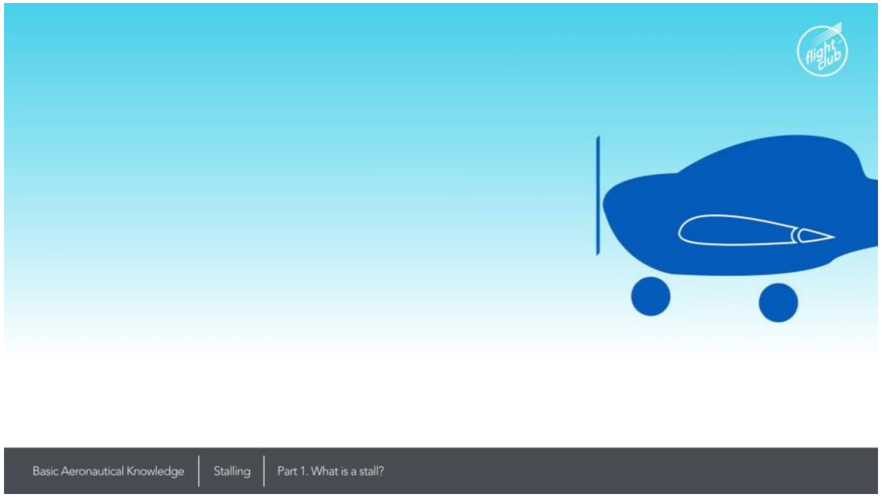


$$C_m = \frac{M}{\frac{1}{2}\rho V_\infty^2 c^2}$$

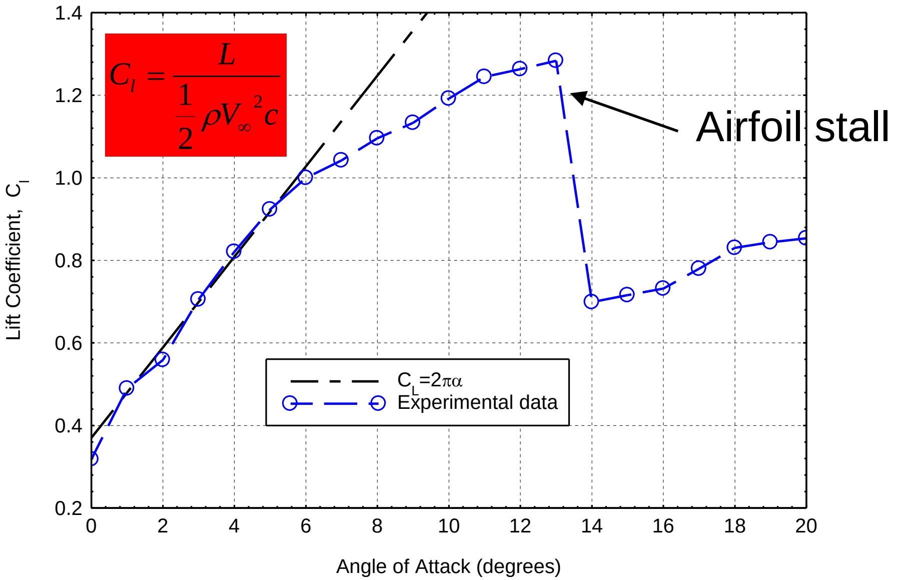
❑ Incompressible Flow Around an Airfoil

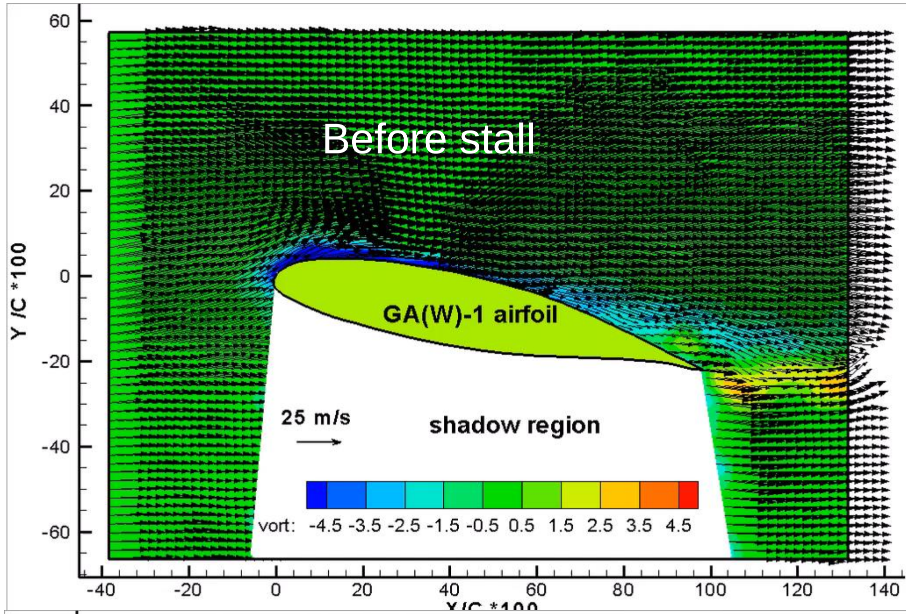


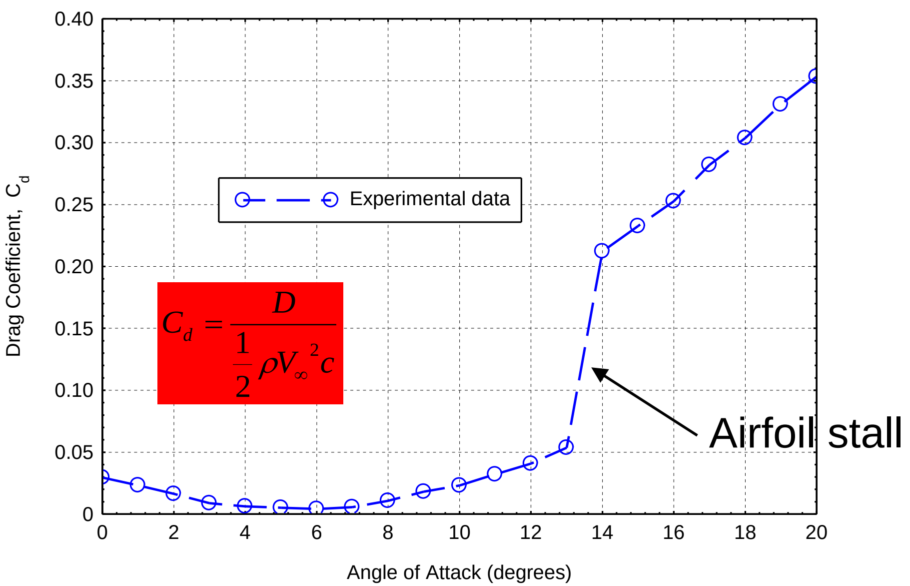


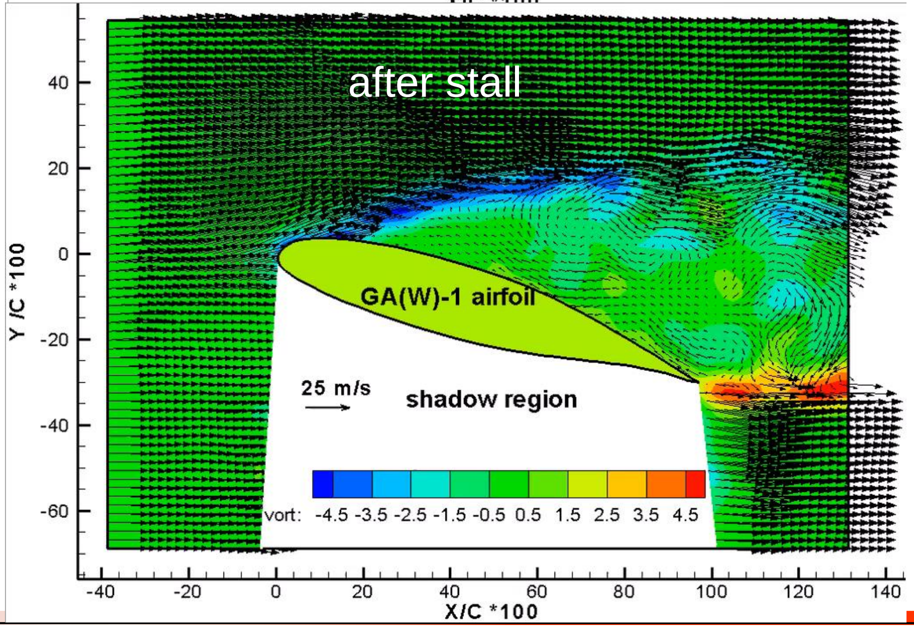


❑ Aerodynamic Performance of an airfoil









❑ How an Airfoil Generates Lift?