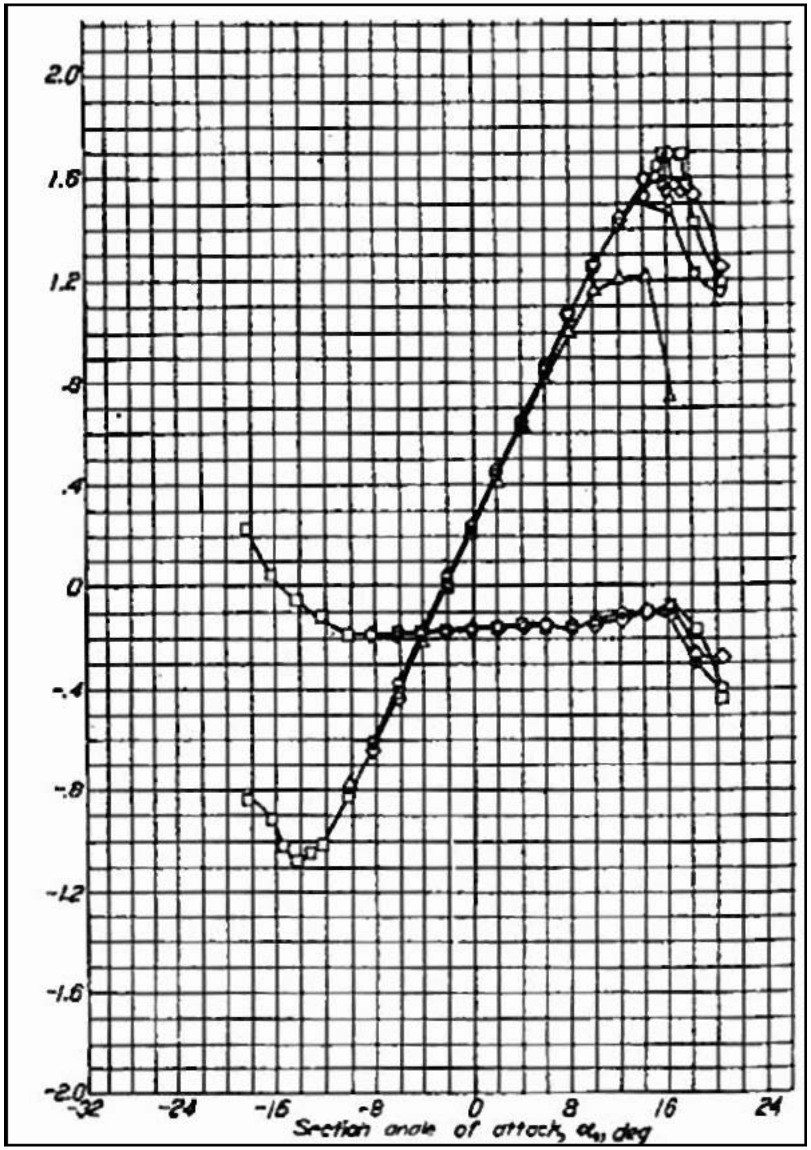
Variation of Cl with the angle of attack and Reynolds



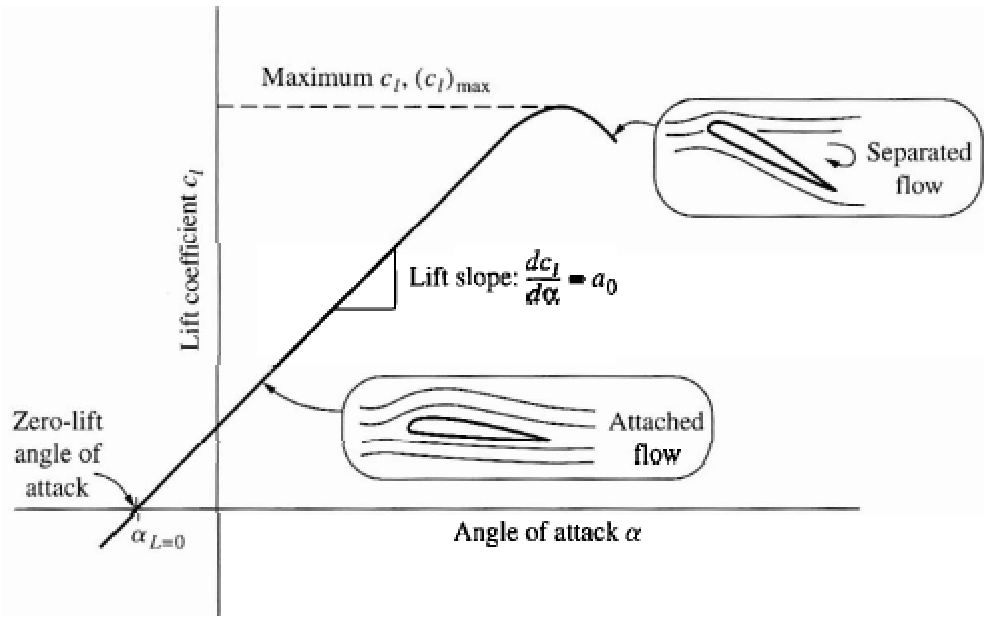
The slope of this linear portion is called the lift slope and is designated by a0. For thin airfoils, a theoretical value for the lift slope is 2pi per radiant, or 0.11 per degree.

there is a finite value of Cl at zero angle of attack, and that the airfoil must be pitched down to some negative angle of attack for the lift to be zero. This angle of attack is denoted by $\alpha_{L=0}$

If positively cambered airfoils have negative zero-lift angles of attack. In contrast, symmetric airfoil has $\alpha_{L=0} = 0$

a negatively cambered airfoil has a positive $\alpha_{L=0}$

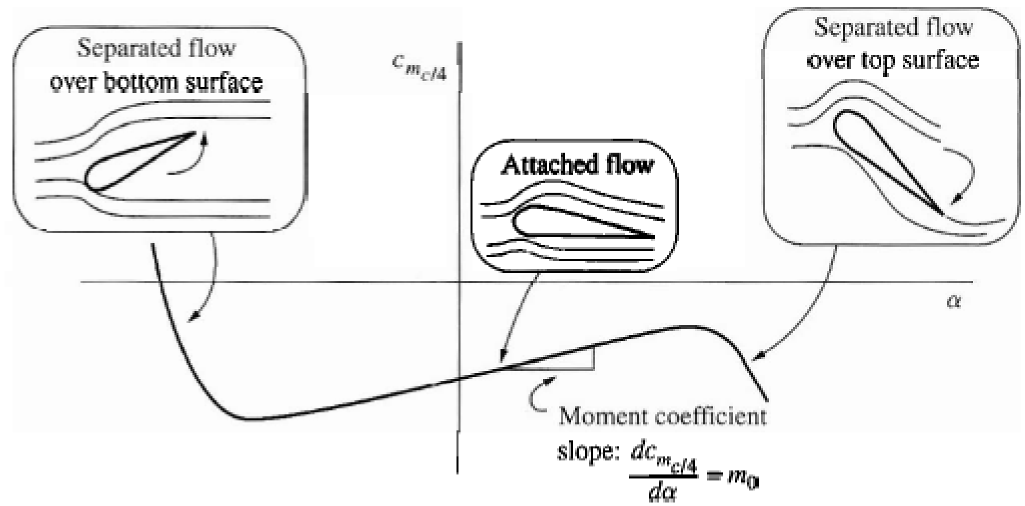
At the other extreme, at high angles of attack, the lift coefficient becomes non-linear, reaches a maximum value denoted by $C_{l\max}$ then drops as a further increased.



This is because a separation occurs over the top surface of the airfoil and the lift decreases (sometimes precipitously). In this condition, the airfoil is said to be stalled. In contrast, over the linear portion of the lift curve, the flow is attached over most of the airfoil surface.

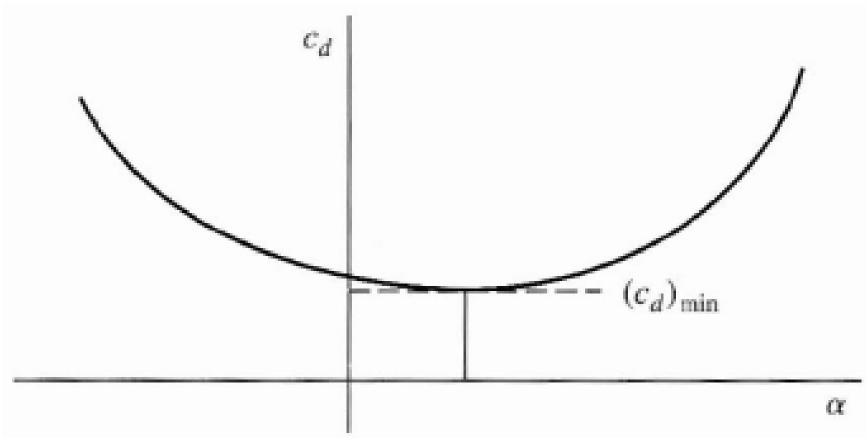
the linear portion of the lift curve is essentially insensitive to variations in Re.
By increasing Reynolds number Clmax increases

Variation of Cm with the angle of attack and Reynolds



over most of the practical range of the angle of attack the slope of the moment coefficient curve is essentially constant.
This slope is positive for some airfoils (as shown here), but can be negative for other airfoils. The variation becomes nonlinear at high angle of attack, when the flow separates from the top surface of the airfoil, and at low, highly negative angles of attack, when the flow separates from the bottom surface of the airfoil.

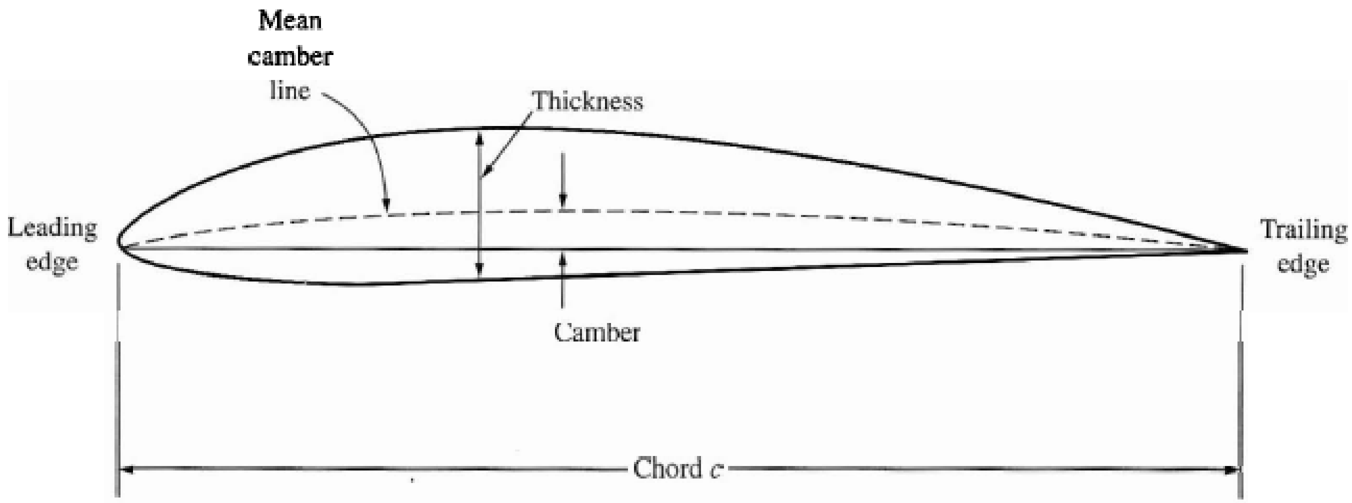
Variation of Cd with the angle of attack and Reynolds



For a cambered airfoil, the minimum value cd does not necessarily occur at zero angle of attack, but rather at some finite but small angle of attack. For this angle-of-attack range, the drag is due to friction drag and pressure drag. In contrast, the rapid increase in cd which occurs at higher values of alpha, is due to the increasing region of separated flow over the airfoil, which creates a large pressure drag.

The friction decreases by increasing the Reynolds number. Moreover, the Reynolds number influences the extent and characteristics of the separated flow region, and hence it is no surprise that *cd* at the larger values of alpha is also sensitive to the Reynolds number.

NACA AIRFOIL NOMENCLATURE



The major design feature of an airfoil is the ***mean camber line,*** which is the locus of points halfway between the upper and lower surfaces, as measured perpendicular to the mean camber line itself. The most forward and rearward points of the mean camber line are the *leading* and *trailing edges,* respectively. The straight line connecting the leading and trailing edges is the *chord line* of the airfoil, and the precise distance from the leading to the trailing edge measured along the chord line is simply designated the *chord* of the airfoil, denoted by *c.* The *camber* is the maximum distance between the mean camber line and the chord line, measured perpendicular to the chord line. The camber, the shape of the mean camber line, and, to a lesser extent, the thickness distribution of the airfoil essentially control the lift and moment characteristics of the airfoil.

First family of airfoils

NACA *airfoils are indicated by a series of 4 digits*. The numbers in the designation mean the following: The first digit gives the maximum camber in percentage of chord. The second digit is the location of the maximum camber in tenths of chord, measured from the leading edge. The last two digits give the maximum thickness in percentage of chord. For example, the **NACA** *2412* airfoil has a maximum camber of *2%* of the chord (or *0.02c),* located at *0.4c* from the leading edge. The maximum thickness is *12%* of the chord (or *0.12c)*

Second family of airfoils

The numbers mean the following: The first digit, when multiplied by *3/2,* gives the design lift coefficient in tenths (the design lift coefficient is defined and discussed in a subsequent paragraph). The second and third digits together are a number which, when multiplied by one-half, gives the location of maximum camber relative to the leading edge in percentage of chord. The last two digits give the maximum thickness in percentage of chord. For example, the **NACA** 23012 airfoil has a design lift coefficient of 0.3, the location of maximum camber at 15% of the chord (or 0.15c) from the leading edge, and a maximum thickness of 12% of the chord (or 0.12c).