

ME 440

Aerospace Engineering Fundamentals

Airfoils

An airfoil's shape is defined by several parameters, which are shown in the figure below.

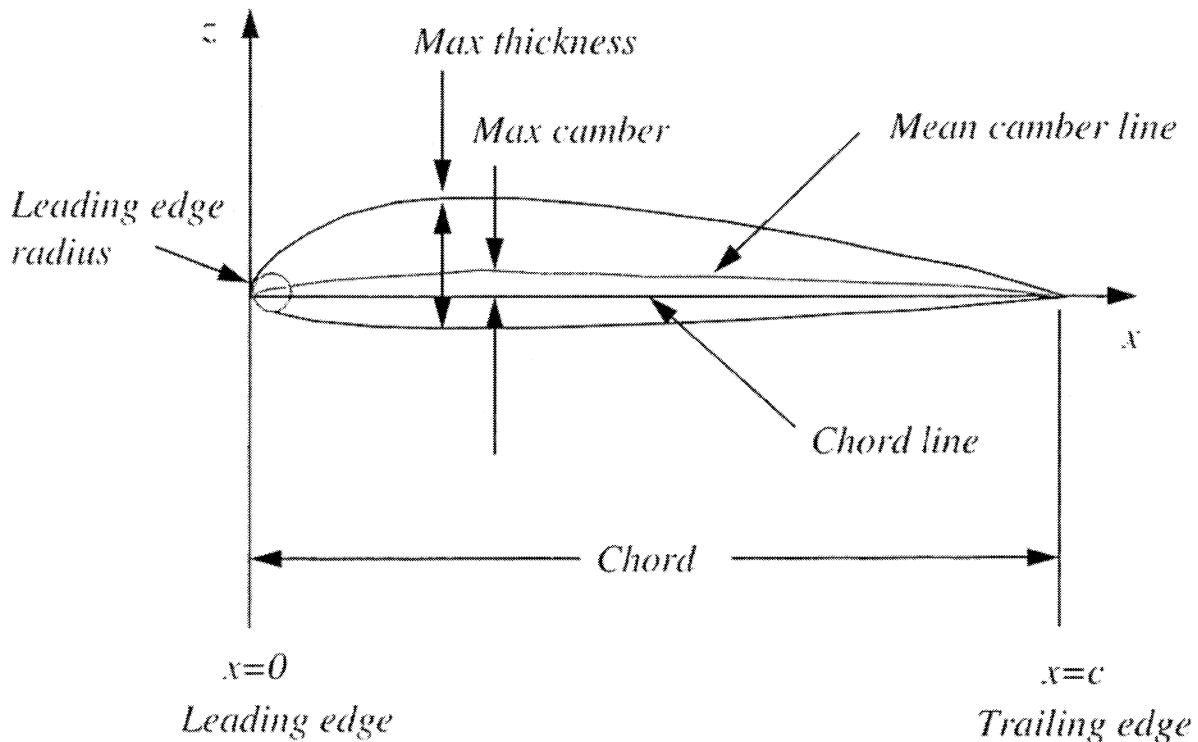


Fig. 3.20 Airfoil shape parameters.

Airfoil Definitions

Chord Line: Straight line drawn from the leading edge to the trailing edge

Chord Length (c): Length of the chord line

Mean Camber Line: Curved line from the leading edge to the trailing edge, which is equidistant between the upper and lower surfaces of the airfoil

Maximum (or Just) Camber: Maximum distance between the chord line and the mean camber line.

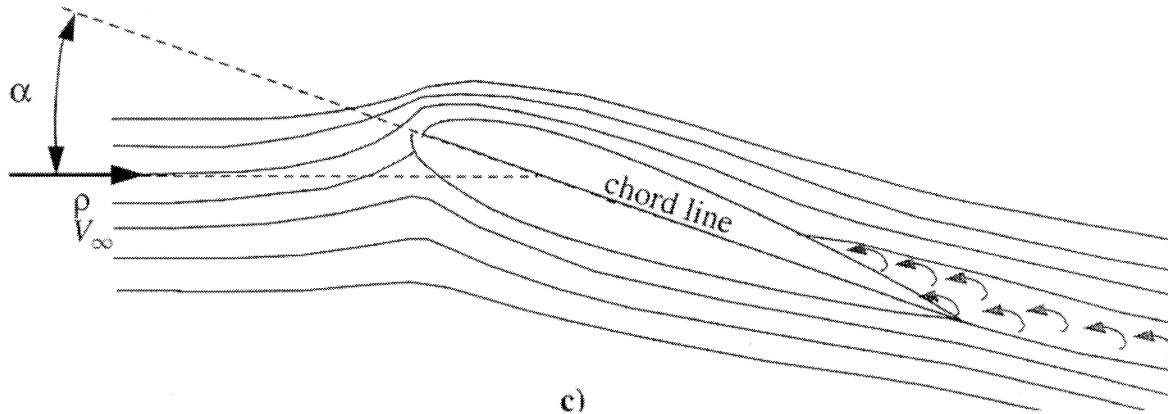
Maximum Thickness: Maximum distance between the upper and lower surfaces of the airfoil normal to the chord line.

Span: Width of the airfoil.

Angle of Attack: Angle between the chord line and the streamwise flow direction.

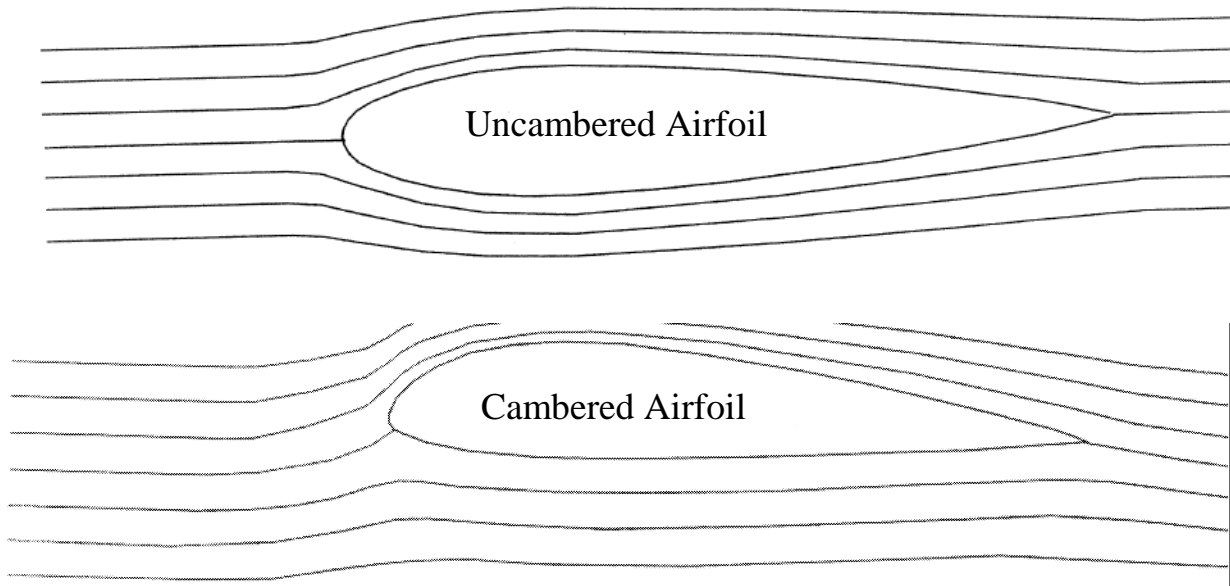
Zero Lift Angle of Attack: Angle of Attack that will produce no lift. For our symmetric wedge this would be an angle of attack of zero.

Stall Angle of Attack: Angle of attack at which there is maximum lift (or lift coefficient)



Symmetric or Uncambered Airfoil: Upper and lower surfaces are mirror images, which leads to the mean camber line to be coincident with the chord line. A symmetric airfoil will also have a just camber of zero.

Cambered Airfoil: An asymmetric airfoil for which the mean camber line will be above the chord line.



Pitching Moment: Torque or moment created on the wing due to net lift and drag forces. Tends to rotate the leading edge either up or down.

Pitching Moment Coefficient:

$$c_m = \frac{m}{\frac{1}{2}\rho V^2 S c}$$

where

m: pitching moment (will depend on the moment reference center)

c: chord length

Center of Pressure: The moment reference center for which the moment is zero. Depends on the angle of attack.

Aerodynamic Center: The moment reference center for which the moment does not vary with angle of attack.

NACA Classification

Airfoils have been classified by the National Advisory Committee for Aeronautics (NACA), the forerunner of NASA, and have been cataloged using a four digit code. Hence a specific airfoil can be identified by NACA WXYZ where

W: maximum camber as % of the chord length

X: Location of the maximum camber form the leading edge along the chord line in tenths of chord length

Y&Z: Maximum thickness in % of the chord length

NACA Airfoil Charts

Every NACA airfoil has two charts to present the lift, drag, and moment coefficient data for the airfoil. The first chart will have curves of lift coefficient versus angle of attack at various Reynolds numbers and curves of moment coefficient at the quarter chord point versus angle of attack at various Reynolds numbers. See the chart below. In addition to the lift and moment coefficients, the stall angle of attack and zero lift angle of attack can be determined.

The second chart will have curves of drag coefficient versus lift coefficient at various Reynolds numbers and curves of moment coefficient at the aerodynamic center versus lift coefficient at various Reynolds numbers. In addition to smooth airfoils, it is common for data for an airfoil whose leading edge has a sandpaper surface texture to be included. The second chart also has an insert picture of the air foil geometry and the aerodynamic center for the airfoil at different Reynolds numbers is provided in tabular form.

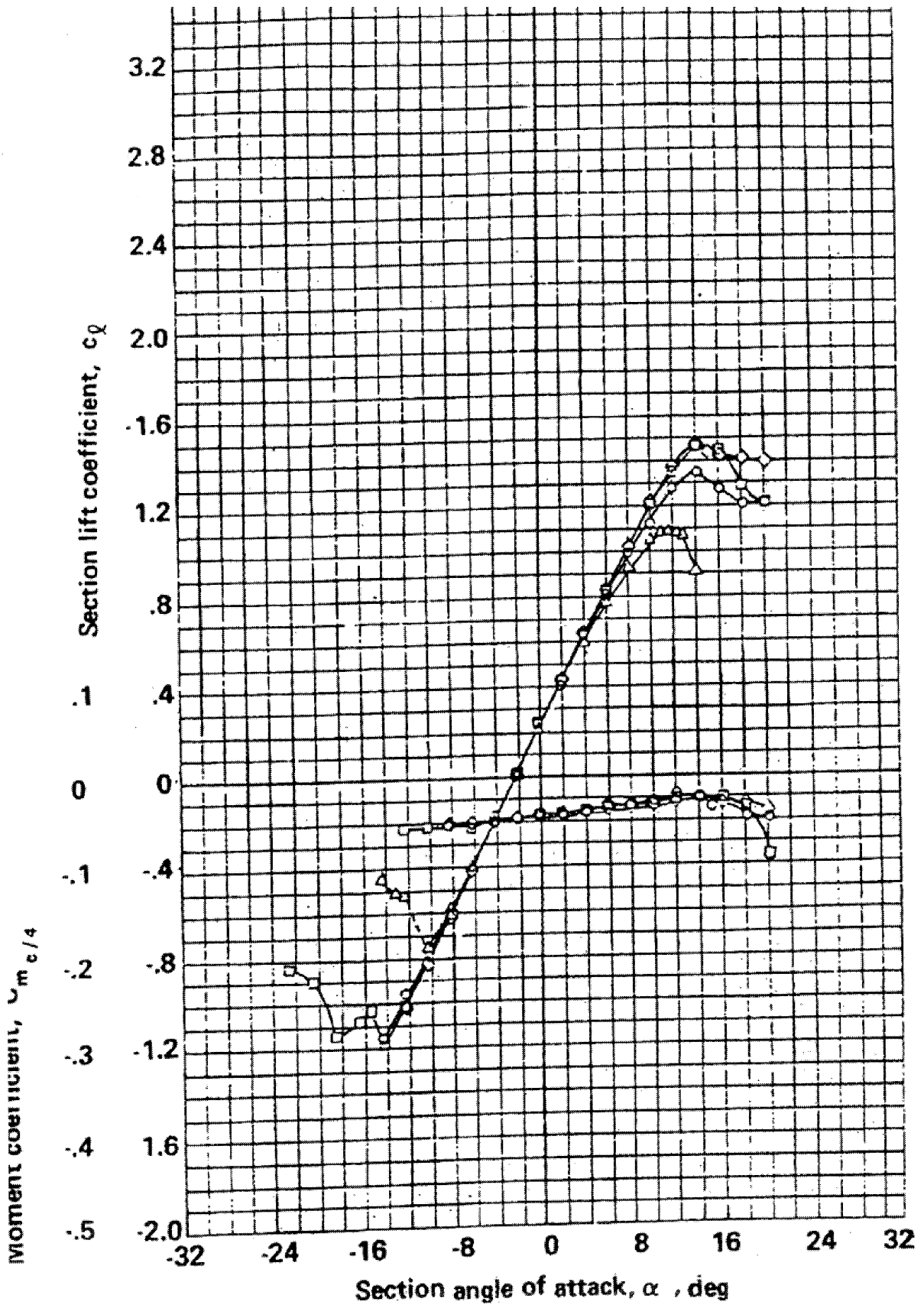


Fig. B.8a NACA 2418.

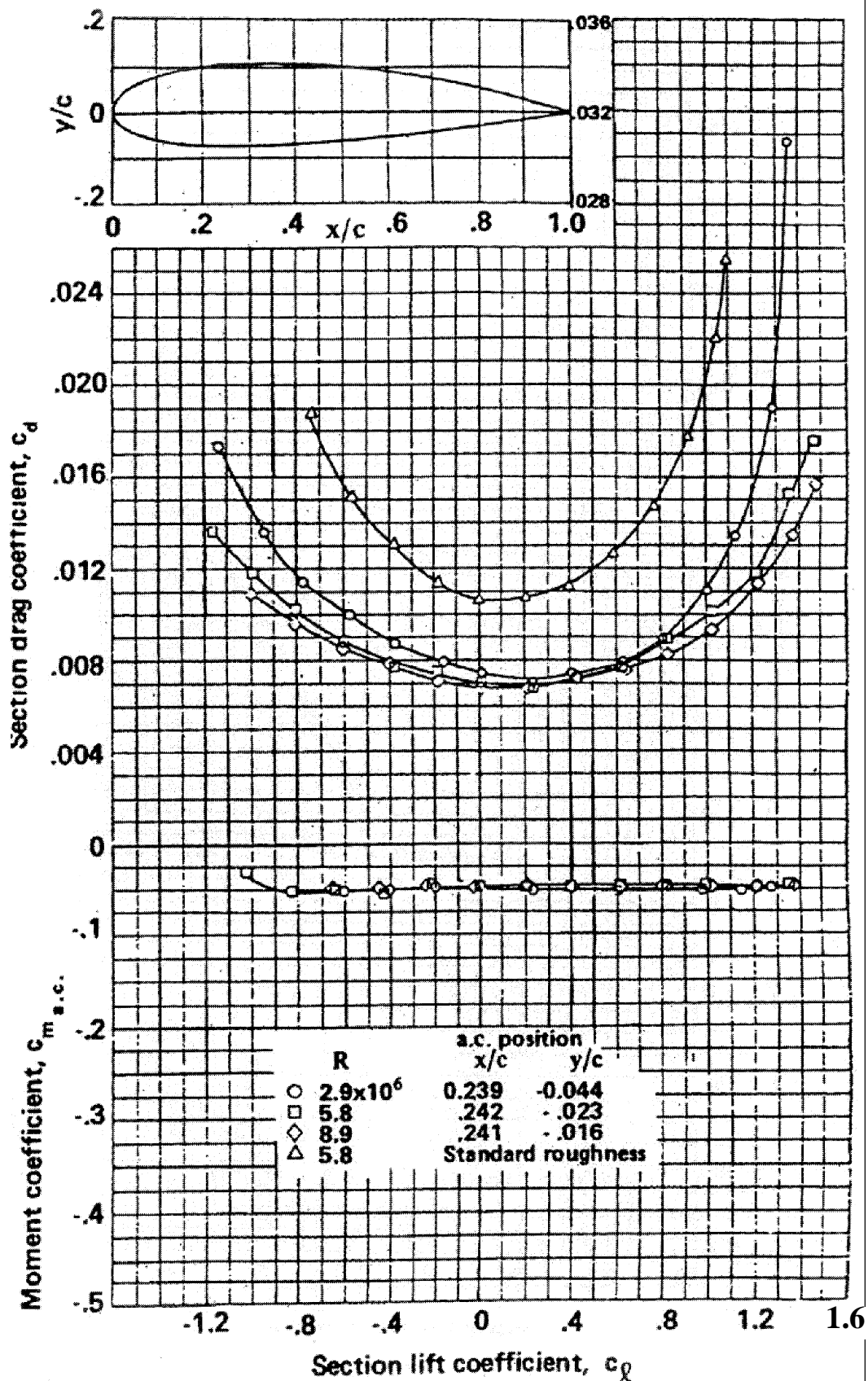


FIG. B.8b NACA 2418.

Using NACA Airfoil Charts

1. Determine the Reynolds number.
2. Using the angle of attack determine the lift coefficient from Chart #1.
3. Using the angle of attack determine the moment coefficient at the quarter chord point from Chart #1.
4. Using the lift coefficient determine the drag coefficient from Chart #2.
5. Using the lift coefficient determine the moment coefficient at the aerodynamic center from Chart #2.
6. Using the Reynolds number determine the airfoil aerodynamic center from the table on Chart #2

Compressibility Effects

For Mach number less than 0.3, we may assume that our flow is incompressible and the standard airfoil charts work very well. For Mach numbers greater than 0.3, we must correct the lift coefficient using the Prandtl-Glauert correction which gives

$$c_L = \frac{c_{L,\text{chart}}}{\sqrt{1 - M_\infty^2}}$$

which is valid for Mach numbers up to 0.7.