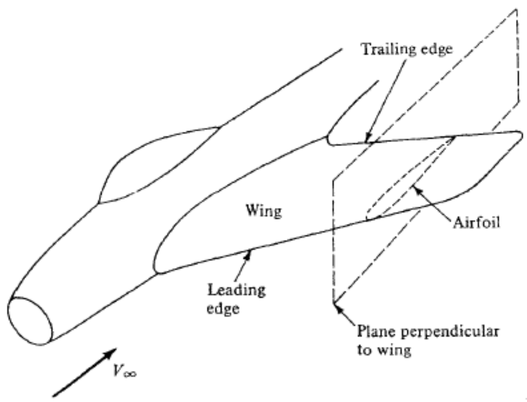
Fundamentals of Aerodynamics Finals

Module 1

Airfoils

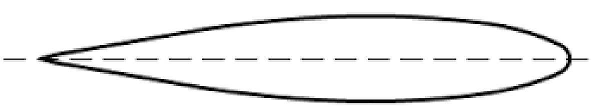
An airfoil or section is the streamlined body which when set at a suitable angle of attack, produces more lift than drag while also producing a manageable pitching moment. It is a solid body designed to move through gaseous medium and obtain useful force reactions other than drag. Example of that is wing, control surface, fin, turbine blade, sail, windmill, blade.



2 types of Airfoils

Symmetrical Airfoil

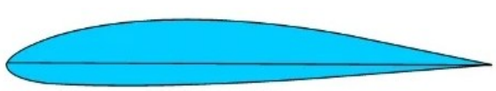
The upper and lower camber are equal in shape



symmetrical airfoil

Asymmetrical Airfoil

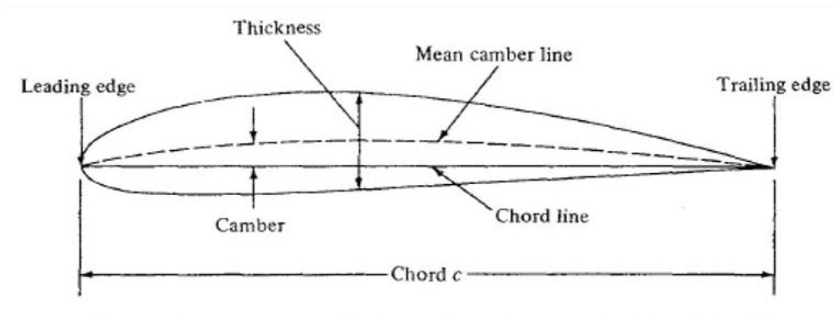
The upper and lower cambers are not equal in shape. The upper camber has a greater curvature than the lower camber.



ASYMMETRICAL AIRFOIL

Topic 2

Airfoil Nomenclature



Chord Line - a straight line connecting the leading and trailing edges of an airfoil.

Mean Camber Line - the locus of points halfway between the upper and lower surfaces.

Chord - the distance between the leading and trailing edges measured along the chord line.

Thickness - the maximum distance between the upper and lower surfaces measured perpendicular to the chord line.

Trailing Edge - the rearmost point of the mean camber line.

Leading Edge - the forward point of the mean camber line.

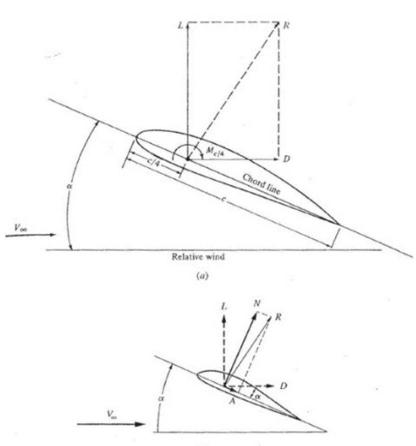
Trailing Edge - the most rearward points of the mean camber line.

Leading Edge - the most forward points of the mean camber line.

Wingspan (b) - the distance from wing tip to wing tip.

Wing Area (S) - the planform area of the wing including ailerons and flaps.

Aspect Ratio (AR) - the ratio of the wingspan to the mean chord. For wings that are not rectangular in shape as viewed from above, the aspect ratio is the ratio of the square of the span to the wing area (AR = b^2/S).



Angles - the angles of attack, incidence, and dihedral are important in the study of airfoil and wing aerodynamics.

Angle of Attack - angle between the relative wind and the chord line.

Aerodynamic Force (R) - created by the pressure and shear stress distributions over the wing surface. This resultant force can be resolved into two forces parallel and perpendicular to the relative wind.

Drag (D) - defined as the component of the aerodynamic force parallel to the relative wind.

Lift (L) - defined as the component of the aerodynamic force perpendicular to the relative wind.

Relative wind – The air far upstream of the airfoil.

Topic 3

NACA AIRFOIL

The NACA airfoil series represents a significant advancement in the field of aerodynamics, developed by the National Advisory Committee for Aeronautics (NACA), the precursor to NASA. These airfoils were designed to optimize the aerodynamic performance of aircraft wings, contributing to the development of more efficient and stable flying vehicles. The NACA airfoil series provided standardized, reliable designs that have remained relevant in both historical and modern aeronautical engineering. The shape is described by a series of digits followed by the word NACA. Can either be 4,5, or 6 digits.

NACA Airfoil Designation

4 – Digit Airfoils

The first digit specifies the maximum camber in percentage of the chord (airfoil length), the second indicates the position of the maximum camber in tenths of the chord, and the last two numbers provide the maximum thickness of the airfoil in percentage of the chord.

Example: NACA – 4412

4 = maximum camber 0.04 (chord)

4 = position of maximum camber 0.4 (chord) from the Leading Edge or L.E

12 = maximum thickness 0.12 (chord)

5 – Digit Airfoils

The first digit specifies the maximum camber in percentage of the chord (airfoil length), the second and third digit indicates the position of the camber in percentage divided by 2, and the last two numbers provide the maximum thickness of the airfoil in percentage of the chord.

Example: NACA – 23012

2 = maximum camber 0.02 (chord)

30 = position of maximum camber (0.03/2) = (0.15) (chord) from the L.E

12 = maximum thickness 0.12 (chord)

Other Formulas in NACA Airfoil

b = wingspan

c = chord

s = wing area

In getting the **chord**, we need to divide the wing area over wingspan **($c = \frac{s}{b}$)**

In getting the **wing area,** we need to multiply the wingspan and chord **(s = bc)**

In getting the **wingspan,** we need to divide the wing area and chord **(b = $\frac{s}{c}$)**

AR - ASPECT RATIO (dimensionless)

The formula for Aspect Ratio → AR = $\frac{b}{c}$ or $\frac{b^2}{s}$

Note: If the **wingspan** is missing and the **Aspect Ratio and Chord** is given, we need to derive the formula:

Derivation of Aspect Ratio:

b = $\sqrt{Ar(s)}$

Example: NACA-4 Digits

Given:

NACA 3112

C = $2m^2$

Find the maximum camber, location of maximum camber, and maximum thickness

Note:

Maximum camber referred to as **Cammax**

Location of maximum camber referred to as **LocCammax**

Maximum thickness referred to as **M.T**

Solution:

Cammax = 3/100 = 0.03 (2) = 0.06m^2

LocCammax = 1/10 = 0.1 (2) = 0.2m^2 away from L.E → do not forget the term "away from L.E" when getting LocCammax

M.T = 12/100 = 0.12 (2) = 0.24m^2

Another Example: NACA-5 Digits

Given:

NACA 33122

Chord: 0.7m^2

Find the maximum camber, location of maximum camber, and maximum thickness

Solution:

Cammax = 3/100 = 0.03(0.7) = 0.021m^2

LocCammax = 31/100 = 0.31÷ 2 = 0.155 (0.7) = 0.1085m^2away from L.E

M.T = 22/100 = 0.22(0.7) = 0.154m^2

Finding the NACA number:

In finding the NACA number, we will just going to reverse the process instead of dividing it by 100 and 10, we will now multiply it and divide each digit using the given chord. The same goes for finding the 5-digit NACA the difference is that we will multiply the given LocCammax by 2 before multiplying it by 100.

Example:

Find the NACA # of 4-digit airfoil

Given:

Cammax = 0.06m^2

LocCammax = 0.2m^2away from L.E

M.T = 0.24m^2

C = $2m^2$

Solution:

Cammax = $\frac{0.06}{2}$ = 0.03(100) = 3

LocCammax = $\frac{0.2}{2}$ = 0.1(10) = 1

M.T = $\frac{0.24}{2}$ = 0.12(100) = 12

NACA 3112

DRAG

The component of aerodynamic force parallel to the relative wind

Formula:

D = $\frac{1}{2}\rho s v^2 C_D$

Where:

D = Drag force in N or lbf

C_D = coefficient of drag (unitless)

ρ = density in kg/m^3 or slugs/ft^3

s = wing area in ft^2 or m^2

v = velocity in fps or mps

LIFT

The component of the aerodynamic force perpendicular to the relative wind

Formula:

L = $\frac{1}{2}\rho s v^2 C_L$

Where:

L = Lift force in N or lbf

C_L = coefficient of Lift (unitless)

ρ = density in kg/m^3 or slugs/ft^3

s = wing area in ft^2 or m^2

v = velocity in fps or mps

Example:

Given:

h = 1.8Km→ 1800m

C_D = 0.14

s = 144m^2

v = 100 m/s

D =?

ρ = ?

Solution:

Use the gradient formula to get the density

T = T_0 + ah

T = 288.2 + (-0.0065) (1800)

T = 276.5k

Solve for Density

ρ = 1.225$(\frac{276.5k}{288.2})^{4.26}$

ρ = 1.0267 $\frac{kg}{m^3}$

We will now compute the Drag

D = $\frac{1}{2}\rho s v^2 C_D$

Substitute every given

D = $\frac{1}{2}(1.0267)(144)(100)^2(0.14)$

D= 103491.3600

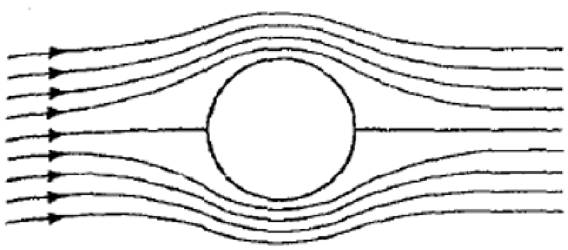
MODULE 2

Topic 1:

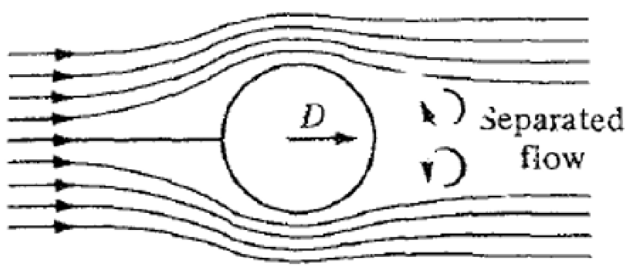
INTRODUCTION TO VISCOUS FLOW

Viscosity refers to a fluid's resistance to deformation and internal friction when in motion. While an **ideal fluid** is often assumed to be **inviscid** (having no viscosity) for simplification in some theoretical models, real-world fluids, such as air and water, always exhibit viscosity, making the study of **viscous flow** essential in aerodynamics.

Comparison between ideal friction flow and real flow with the effects of friction on a sphere:



Frictionless flow: no drag



Real flow: finite drag

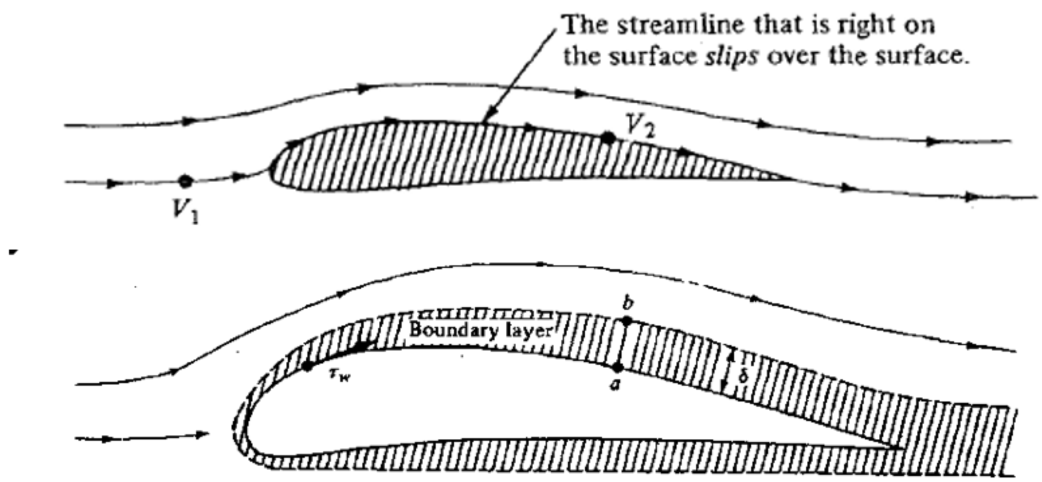
Frictionless Flow:

The study of ideal frictionless flow around a sphere provides valuable theoretical insights into fluid dynamics, though it deviates significantly from reality. The assumptions of **symmetrical streamlines, no boundary layer, no flow separation, and zero drag** help simplify calculations and establish fundamental aerodynamic principles. However, real-world applications require the consideration of **viscous effects**, as they play a crucial role in determining flow behavior, drag forces, and overall aerodynamic efficiency.

Real Flow:

In real-world fluid dynamics, viscosity significantly alters the flow around a sphere by introducing frictional effects. Unlike an idealized model where fluid moves without resistance, real flow develops a boundary layer—a thin region near the sphere's surface where velocity gradually increases from zero (due to the no-slip condition) to the free-stream velocity. As the fluid moves past the sphere, an adverse pressure gradient on the backside causes flow separation, leading to a turbulent wake and increased aerodynamic drag. This drag arises from two main factors: skin friction drag, caused by shear stress within the boundary layer, and pressure drag, resulting from the low-pressure wake behind the sphere. These viscosity-induced effects are crucial in aerodynamics, influencing the design of aircraft, vehicles, and sports equipment.

Frictionless flow in an airfoil:



Boundary layer: The region of viscous flow which has been retarded owning to friction at the surface. A flow field can be split into two regions, one region in which friction is important, namely in the boundary layer near the surface, and another region of frictionless flow (sometimes called *potential flow)* outside the boundary layer. The boundary thickness grows as the flow moves over the body; that is more and more of the flow is affected by friction as the distance along the surfaces increases. In addition, the presence of friction creates a shear stress at the surface. This shear stress has dimensions of force/area and acts in a direction tangential to the surface. This give rise to a drag force called **skin friction drag**

Absolute coefficient of viscosity

For liquids **μ decreases** as **T** increases (we all know that oil gets thinner when temperature is increased). But for gases, **μ** increases as **T** decreases (air gets thicker when temperature is decreased).

For air at standard sea-level temperature

For SI:

μ= $1.7894x10^{-5}\,\frac{kg}{m*s}$

For English:

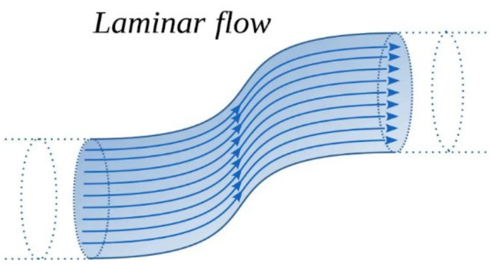
μ= $3.7373 \text{x} 10^{-7} \frac{slugs}{ft*s}$

Topic 2:

Difference between Laminar flow and Turbulent flow

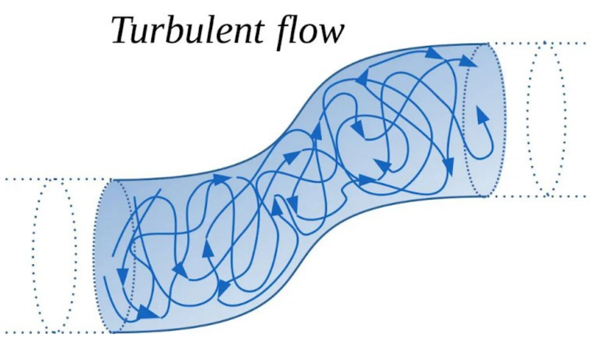
Laminar Flow

It is the flow in which the streamlines are smooth and regular, and the fluid element moves smoothly along the streamline and note the Reynolds number will be less than 499,999



Turbulent Flow

Flow in which the streamline breakup and fluid element moves in a random, irregular, and tortuous fashion and the Reynolds number will be greater than 500,000



Topic 3:

Reynold's number

In honor of Osborne Reynolds (1842-1912). A number of importance and impact on Aerodynamics equal to those of the Mach number. It is the measure of the ratio of inertia forces to viscous forces. Reynolds number is a similarity parameter that helps determine whether the flows in a body and its scaled version are aerodynamically similar. It can be applied in determining whether all, or portion of the boundary layer is Laminar or Turbulent.

Formula for Reynolds Number:

$$\boldsymbol{Re_x = \frac{\rho V x}{\mu}}$$

Where:

$\boldsymbol{\rho}$ = Density of the fluid ($\frac{kg}{m^3}$)

$\boldsymbol{V}$ = Velocity of the fluid relative to the object ($\frac{m}{s}$)

X = Chord, LocCammax, Chord length

Sample problem:

Given:

Chord Length: 5cm long → 0.005m

ρ = 1.225kg/m^3

v = 120m/s

μ = 1.7894x10^{-5}

Find the Reynolds Number and determine what type of flow

Solution:

$$Re_x = \frac{\rho V x}{\mu}$$

Substitute the givens

$$Re_x = \frac{(1.225)(120)(0.005)}{1.7894x10^{-5}}$$

Re_x = 41,075.22074 Laminar flow

Critical Reynold Number for transition layer

The **critical Reynolds number** is the value at which the transition from laminar to turbulent flow begins. This transition does not occur at a single precise Reynolds number but rather over a range of values, depending on several factors such as surface roughness, pressure gradients, and external disturbances. In general, for flow over a flat plate, transition typically begins at a Reynolds number of **approximately 5 × 10^5 (500,000)** based on the distance from the leading edge.

Formula for Critical Reynolds

$$Re_{xcr} = \frac{\rho V x_{cr}}{\mu}$$

X_{cr} = Critical point

Example:

Given:

ρ = 1.225kg/m^3

v = 120m/s

μ = 1.7894x10^{-5}

X_{cr} = 1

Find the Critical Reynolds and determine what type of flow

Solution

$$Re_{xcr} = \frac{\rho V x_{cr}}{\mu}$$

Substitute every given

$$Re_{xcr} = \frac{(1.225)(120)(1)}{1.7894x10^{-5}}$$

Re_{xcr} = 8899631.161 turbulent