Module – 2

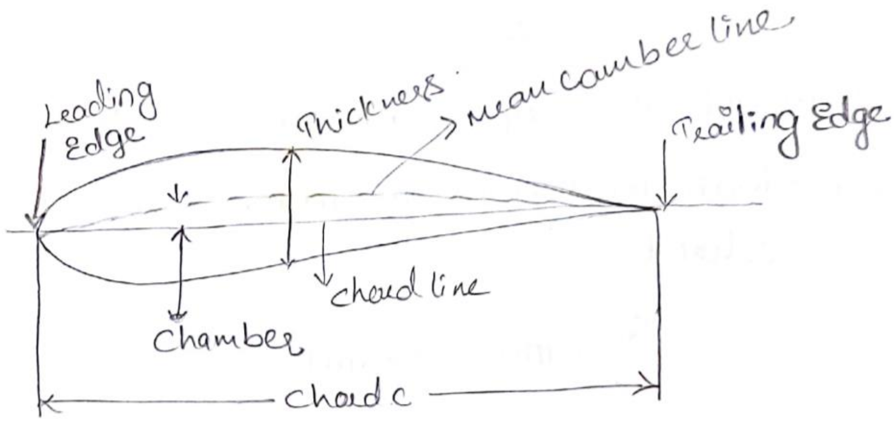
BASIC PRINCIPLES OF FLIGHT

Syllabus:

Basic principles of flight – significance of speed of sound; airspeed and groundspeed; standard atmosphere; Bernoulli's theorem and its application for generation of lift and measurement of airspeed; forces over wing section, aerofoil nomenclature, pressure distribution over a wing section.Lift and drag components – generation of lift and drag; lift curve, drag curve, types of drag, factors affecting lift and drag; centre ofpressure and its significance; aerodynamic centre, aspect ratio, Mach number and supersonic flight effects; simple problems on lift and drag.

1. Write in detail about the aerofoil nomenclature with sketch.
Or
With a neat sketch define nomenclature of asymmetric aerofoil

Airfoil nomenclature



Airfoil :- The cross-sectional shape obtained by the intersection of the wing with the perpendicular plane is called an airfoil.

Mean Camber Line :- The locus of points halfway between the upper and lower ~~section~~ Surfaces are called Mean camber line.

Leading Edge :- The most forward points of the mean camberline.

Trailing Edge :- The most rearward points of the mean camberline.

Chord line :- The straight line connecting the leading and trailing edges is called the chord line is also designated as chord 'c'.

Camber :- The maximum distance between the mean camber line and chord line.

2. Explain about: Aerodynamic Centre and Centre of Pressure.

Centre of Pressure and its significance

The wing is pushed or pulled through the air at a small angle called the angle of attack or angle of incidence. we hold it at the centre of each end, then not only shall we feel an upwards and backwards force exerted upon it, but it will tend to rotate, its leading edge going over the top.

If we try to make it glide of its own accord, it will turn over and over. This is because the effective or resultant force acting upon it is in front of the centre line, when it is left free to fly by itself, its weight is acting downwards at the centre.

if we add weights to it so that its centre of gravity is farther forward, it tends to turn the other way, the nose dipping downwards. That position is found as centre of pressure

Aerodynamic center

The aerodynamic center is the point at which the pitching moment coefficient for the airfoil does not vary with lift coefficient (angle of attack).

$$\frac{dC_m}{dC_L} = 0.$$ (where C_L is the aircraft lift coefficient).

3. Explain how a pressure distribution over a wing section occurs.

Pressure distribution over wing section

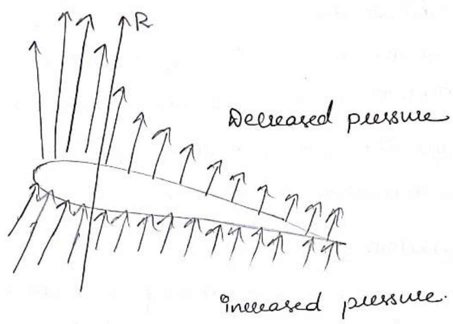
Pressure distribution over wing section

The top surface of the wing is in curving the air flow downwards; the bottom surface acts in much the same way on both the airfoil shape and on the flat plate.

The single force acting at a place called the centre of pressure but it is in reality the sum total of all the pressure acting upon the surface of the airfoil, this pressure being distributed all over the surface.

Consider an airfoil board having a small holes round the wing are connected to glass tubes, or manometers, in which there is a column of liquid, the glass tubes being connected at the bottom to a common reservoir. If the liquid in any tube is sucked upwards, it means that the pressure at the corresponding hole on the surface of the wing has been reduced; similarly, if the liquid is forced downwards, the pressure has been increased.

As per this the draw a pressure distribution of the airfoil. First, that over most of the top surface the pressure is decreased – this is due to the downward curvature of the air; on the bottom surface, however, the air is pressed downwards, and there is an increase of pressure.



The decrease in pressure on the top surface is much more marked than the increase underneath, and thus the top surface contributes the largest proportion of the lift.

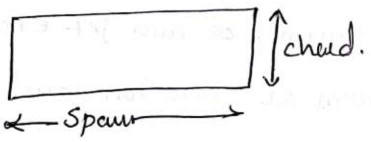
The majority of the lift, both comes from the front portion of the wing, and therefore when we replace all this distributed pressure by a single force, that force as acting in front of the centre of the aerofoil. this is called the centre of pressure.

4. Discuss: Aspect Ratio, Mach Number and Supersonic Flight Effects.

Aspect ratio

aspect ratio:-

The ratio of its span to its chord is called the aspect ratio of the wing.



$$\text{Aspect ratio} = \frac{\text{span}}{\text{chord}}.$$

$$AR = \frac{b^2}{S}.$$

The aspect ratio AR is the ratio of the square of the wingspan b to the projected wing area S, which is equal to the wingspan b to the mean aerodynamic chord.

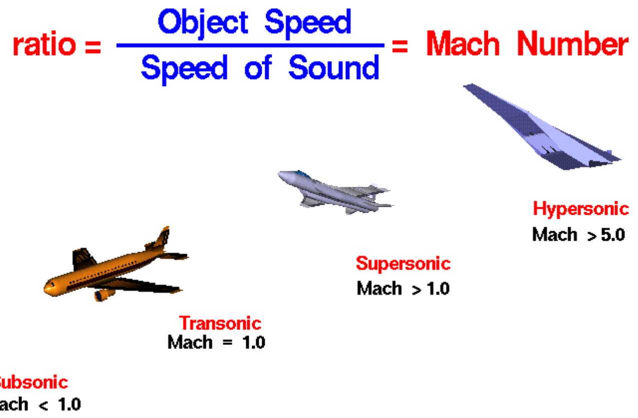
Mach number

The speed of aeroplanes in relation to the speed of sound and to say that they are travelling at half, or three-quarters of the speed of sound.

i.e., Mach no of 0.5 simply meaning that the aeroplane is travelling at half the speed of sound.

It is the ratio of object speed ~~divi~~ to the speed of sound.

$$M = \frac{u}{c} = \frac{\text{object speed}}{\text{speed of sound}}.$$



Supersonic flight effects

* The Drag is of course very high, which means that great thrust is required to maintain flight.

* The propeller is not a practical proposition at these speeds, and we have to rely on the Jet Engine or ram-jet etc..

* The controls are too heavy for manual operation, and power driven controls become a virtual necessity.

* The control surfaces themselves are not so effective as in subsonic flight. because

  (i) They are working in that part of the air flow which tends to separate from the surface.

  (ii) They do not have any effect on the flow over the surface in front of them.

* Mach-2, The rise in temperature caused by the motion of the aeroplane through the air and the consequent skin friction

5. Define with equation speed of sound. Explain its significance in determining airspeed and ground speed.

Speed of Sound

The speed of sound is the distance travelled per unit time by a sound wave as it propagates through an elastic medium.

The speed of sound in an ideal gas depends only on its temperature and composition. The speed has a weak dependence on frequency and pressure in ordinary air, deviating slightly from ideal behavior.

Speed of Sound Equation, $a = \sqrt{\gamma R T}$

Significance of Speed of Sound to Determine Ground Speed and Airspeed

Aerodynamic forces are generated by an object moving through a fluid (liquid or gas). A fixed object in a static fluid does not generate aerodynamic forces. To generate lift, an object must move through the air, or air must move past the object. Aerodynamic lift depends on the square of the velocity between the object and the air. Now things get confusing because not only can the object be moved through the air, but the air itself can move.

To properly define the relative velocity, it is necessary to pick a fixed reference point and measure velocities relative to the fixed point. The reference point is fixed to the ground, but it could just as easily be fixed to the aircraft itself. It is important to understand the relationships of wind speed to ground speed and airspeed.

Wind Speed: For a reference point picked on the ground, the air moves relative to the reference point at the wind speed.

Ground Speed: For a reference point picked on the ground, the aircraft moves relative to the reference point at the ground speed.

Airspeed: The important quantity in the generation of lift is the relative velocity between the object and the air, which is called the airspeed. Airspeed cannot be directly measured from a ground position, but must be computed from the ground speed and the wind speed.

Mach number, however, is measured with respect to the velocity of the object through the air, or its airspeed.

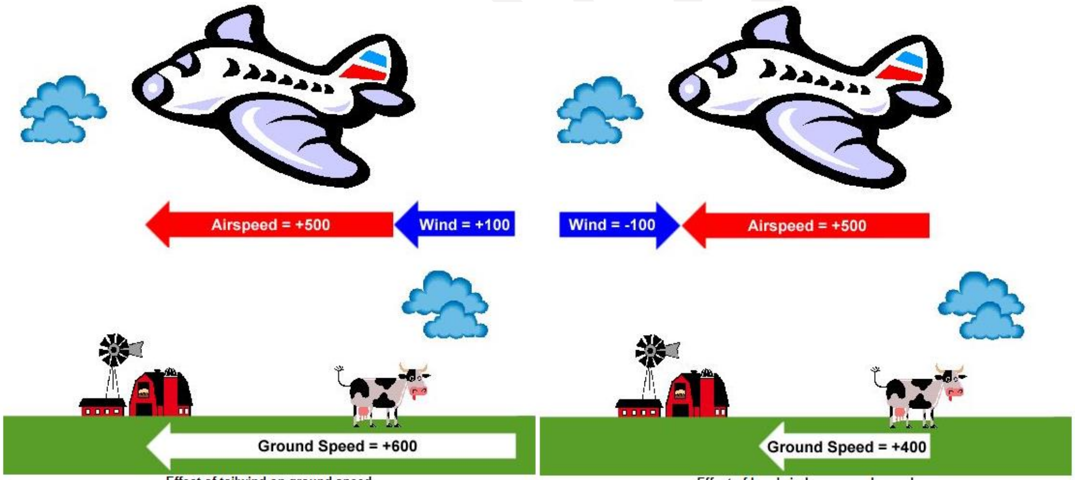
The relationship between ground speed, airspeed, and wind speed can be written in the following form:

$$V_{ground} = V_{airspeed} + V_{wind}$$

If the wind moves in the same direction as the aircraft, the term V_{wind} will have a positive sign. If the wind blows in the opposite direction as the plane's motion, V_{wind} will be a negative number.

The airspeed can be sub divided again into four different levels:

- **Indicated airspeed:** Speed of aircraft as shown on pilot as indicated airspeed.
- **Calibrated airspeed:** It is the indicated airspeed of an aircraft corrected for both position and instrument error.
- **Equivalent airspeed:** It is calibrated airspeed of aircraft correct for adiabatic compressible flow for particular altitudes.
- **True airspeed:** Speed of an aircraft relative to undisturbed air is called true airspeed. It is equivalent airspeed by square root of density ratio.



Effect of tailwind on ground speed

Effect of headwind on ground speed

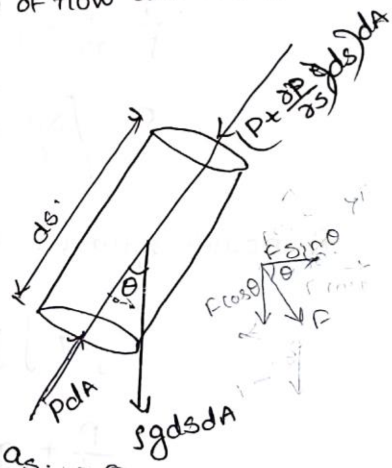
6. Derive Bernoulli's equation and explain its significance in generation of lift and measuring airspeed.

Bernoulli's theorem

Consider a stream line in which flow is taking place in s-direction. Consider a cylindrical element of cross-section dA and length ds. The forces acting on the cylindrical element are

1. Pressure force PdA in the flow direction
2. pressure force $\left(p+\frac{\partial P}{\partial s}ds\right)dA$ opposite to the flow direction.
3. weight of element $\rho g\, dA\, ds$.

Let θ is the angle b/w the direction of flow and the line of action of the weight of element.



Resultant force on the fluid element = mass of element × acceleration

$$F = m \times a$$

$$PdA - \left(P+\frac{\partial P}{\partial s}ds\right)dA - \rho g\, ds\, dA\cos\theta = \rho\, ds\, dA \times a_s \rightarrow (1)$$

where $a_s = \frac{dV}{dt}$

$$= \frac{\partial V}{\partial s}\cdot\frac{ds}{dt} + \frac{\partial V}{\partial t} = V\frac{\partial V}{\partial s} + \frac{\partial V}{\partial t}$$

$\left[\because \text{ steady flow } \frac{\partial V}{\partial t} = 0.\right]$

$$a_s = \frac{V\partial V}{\partial s}.$$

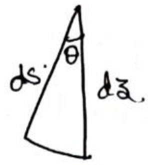
Substitute ~~abo~~ a_s value in Eq (1).

$$\cancel{P}dA - \cancel{P}dA - \frac{\partial P}{\partial s}ds\,dA - \rho g\,ds\,dA\cos\theta = \rho\,ds\,dA \times a_s.$$

$$-\frac{\partial P}{\partial s} ds dA - \rho g ds dA \cos\theta = \rho dA ds \times \frac{\partial V}{\partial s} V$$

Divide $\rho ds dA$

$$-\frac{\partial P}{\partial s \rho} - g\cos\theta = \frac{\partial V}{\partial s} V$$



$$\frac{\partial P}{\rho \partial s} + g\cos\theta + \frac{\partial V}{\partial s} V = 0.$$

where $\cos\theta = \frac{dz}{ds}$.

$$\frac{1}{\rho}\frac{\partial P}{\partial s} + g\frac{dz}{ds} + \frac{V \partial v}{\partial s} = 0$$

or $\boxed{\frac{\partial P}{\rho} + g dz + v\, dv = 0}$ ⟹ Eulers Equation.

Bernoullis Equation is applied while using integration

$$\int \frac{\partial P}{\rho} + \int g dz + \int v dv = \int 0.$$

$$\frac{P}{\rho} + gz + \frac{v^2}{2} = \text{const.}$$

$$\frac{P}{\rho g} + z + \frac{v^2}{2g} = \text{const}$$

$\frac{P}{\rho g}$ → Pressure Energy

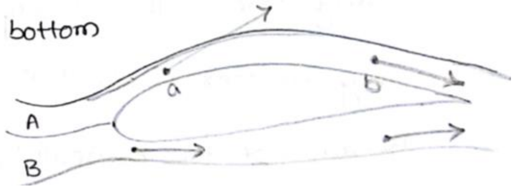
$v^2/2g$ - Kinetic Energy.

z → potential Energy.

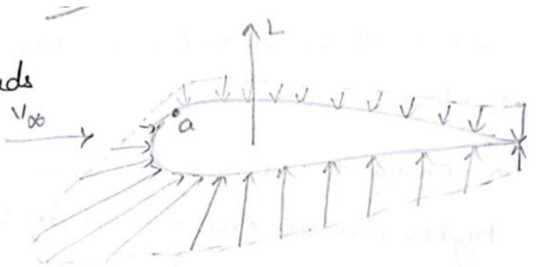
Statement: It states that in a steady, ideal flow of an incompressible fluid, the total energy at any point of the fluid is const.

Bernoulli's theorem application for generation of lift

Consider the flow over an aerofoil. Consider the stream tubes. A + B. The stream tube A flows over the top surface, and the stream tube B flows over the bottom surface. Both stream tubes originate in the free stream



ahead of the airfoil.



As stream tube A flows towards the airfoil, it senses the upper portion of the airfoil as an obstruction, and A must move out of the way of this obstruction.

Stream tube A is squashed to a smaller cross-sectional area as it flows over the nose of the airfoil. The velocity of the flow in the stream tube must increase where it is squashed.

point a long arrow indicates that the higher velocity, then downstream, its cross-sectional area gradually increases and the flow velocity decreases ($\rho AV = const$). Such as shown in a point b.

Now consider stream tube B, which follows the bottom surface of the airfoil. The bottom surface of the airfoil presents less of an obstruction to stream tube B, and so it is not squashed as much as A. The flow velocity in stream tube B is less than B.

* For an incompressible flow, from Bernoulli's Equation.

$$P + \frac{1}{2}\rho V^2 = const$$

clearly where the velocity increases, the static pressure decreases

Similarly for compressible flow. $dp = -\rho V dV$.

Because of the Bernoulli's effect, the pressure over the top surface of the airfoil is less than the pressure over the bottom surface.

* owing to the lower pressure over the top surface and the higher pressure over the bottom surface, the airfoil feels a lift force in the upward direction.

Bernoulli's theorem application for measurement of airspeed (Dec/Jan 2017)

Consider ~~P~~ the Bernoulli's Equation

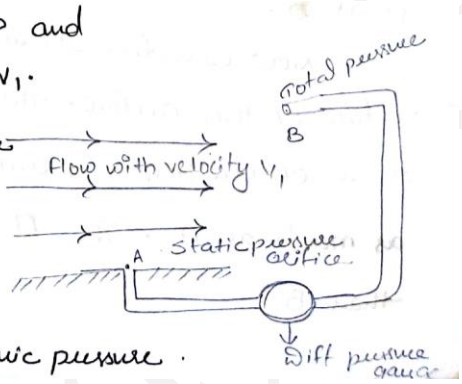
$$\frac{P}{\rho g} + z + \frac{V^2}{2g} = \text{const}$$

$$P + \rho g z + \rho \frac{V^2}{2} = \text{const}$$

$$P + \rho \frac{V^2}{2} = \text{const} \quad \text{for static, level of flow.}$$

Now At point A, the pressure is P and velocity is V_1.

At point B, the pressure is P_0 and the velocity is zero.



$$P + \rho \frac{V_1^2}{2} = P_0 + 0$$

~~where~~ Then $q = \frac{1}{2}\rho V^2$ – dynamic pressure.

$q = \frac{1}{2}\rho V^2$ – dynamic pressure for ~~flo~~ all types of flows.

~~P6*~~ $P + q = P_0$.

For incompressible flow:

$$P + \frac{1}{2}\rho V^2 = P_0$$

$$\frac{1}{2}\rho V^2 = P_0 - P$$

$$V^2 = \frac{2(P_0 - P)}{\rho}$$

$$\boxed{V = \sqrt{\frac{2(P_0 - P)}{\rho}}}$$

The true airspeed of the airplane is:

$$V_{true} = \sqrt{\frac{2(P_0 - P)}{\rho}}$$

Equivalent airspeed $V_e = \sqrt{\frac{2(P_0 - P)}{\rho_s}}$

7. Problems

Q.1) An aircraft carries 40,000 lbs, wing area of 350 ft^2 and wing span of 50fts. At sea level the aircraft flies at 200 and 600ft/sec. What are the values of induced drag and total drag coefficient? Also calculate coefficient of lift when lift = weight. Assume e = 0.85.

$W = 40{,}000\,lbs = 18143.6948\,kg$

1 lb = 0.4536 kg
1 ft = 0.3048 m
1 ft^2 = 0.092
1 ft/sec = 0.305 m/sec

wing area $S = \cancel{30}350\,ft^2$

$= \cancel{106.68\,m^2}$

$= 32.5161\,m^2$

wing span $b = 50\,fts = 15.24\,m$

altitude 200 ft/sec to 600 ft/sec.

induced drag $D_i = ?$ + Lift = ?

$D_{tot} = ?$

e = 0.85 at Lift = weight

Lift = weight.

$L = W$

$\rho_\infty = 1.225\,kg/m^3$

$q_\infty S C_L = W$

$C_L = \dfrac{W}{q_\infty S}$

co-efficient of Lift at velocity 200ft/sec

$V_\infty = 200 \times 0.305\,m/sec$

$= 61\,m/sec$

$q_\infty = \frac{1}{2}\rho_\infty V_\infty^2$

$= \frac{1}{2} \times 1.225 \times (61)^2$

$= 2279.1125\,kg\text{-}m/sec^2$

$C_L = \dfrac{18143.6948}{2279.1125 \times 32.5161}$

$= 0.245$

co-efficient of Lift at velocity $V_\infty = 600 \times 0.305$

$= 183\,m/sec$

$q_\infty = \frac{1}{2} \times 1.225 \times (183)^2$

$= 20512.01\,kg\text{-}m/sec^2$

$C_L = \dfrac{18143.6948}{20512.01 \times 32.5161}$

$C_L = 0.027$

induced drag $D_i = q_\infty S C_{di}$

$$C_{di} = \frac{C_L^2}{\pi e AR}$$

$$AR = \frac{b^2}{S}$$

$$= \frac{(15.24)^2}{32.5161}$$

$$= 7.143$$

$$= \frac{(0.027)^2}{\pi \times 0.85 \times 7.143}$$

$$= 3.82 \times 10^{-5}$$

$$D_{tot} = D_p + D_i$$

$$= C_{Pd} q_\infty S + q_\infty S C_{di}$$

$$= q_\infty S (C_{Pd} + C_{di})$$

$$= 20512.01 \times 32.5161 (C_{Pd} + 3.82 \times 10^{-5})$$

$$D_{tot} = 6.67 \times 10^{-5} C_{Pd} + 25.49$$

8. Discuss about the distribution of temperature in the standard atmosphere.

Temperature related layers:

Troposphere (0 km to 11 km):

- First layer and extend up to 8km near the poles, 11km in mid latitudes and 16km equator.
- Atmosphere is most dense in this region.
- Temperature decreases from 15ºc sea level to -56ºc and this point is called tropopause.
- It contains almost all atmospheric water, 90% of air mass.

Stratosphere (11 km to 51 km):

- Starts just above the troposphere and extend up to 51km from earth surface.
- Temperature is constant for 1st 20km (-56ºc)
- It increases at 1ºc /km up to 32km.
- It increases at 2.8ºc /km up to 47km.
- Ozone layer lies in this layer.
- Stratopause separates stratosphere from mesosphere.

Mesosphere (51 km to 85 km):

- It just lies above stratosphere and extend up to 85km high.
- Temperature decreases @ rate of -2.8ºc from 71km to 85km.
- Temperature decreases @-2.8ºc from 51km to 71km.
- Mesopause separates the layers of mesosphere and thermosphere.

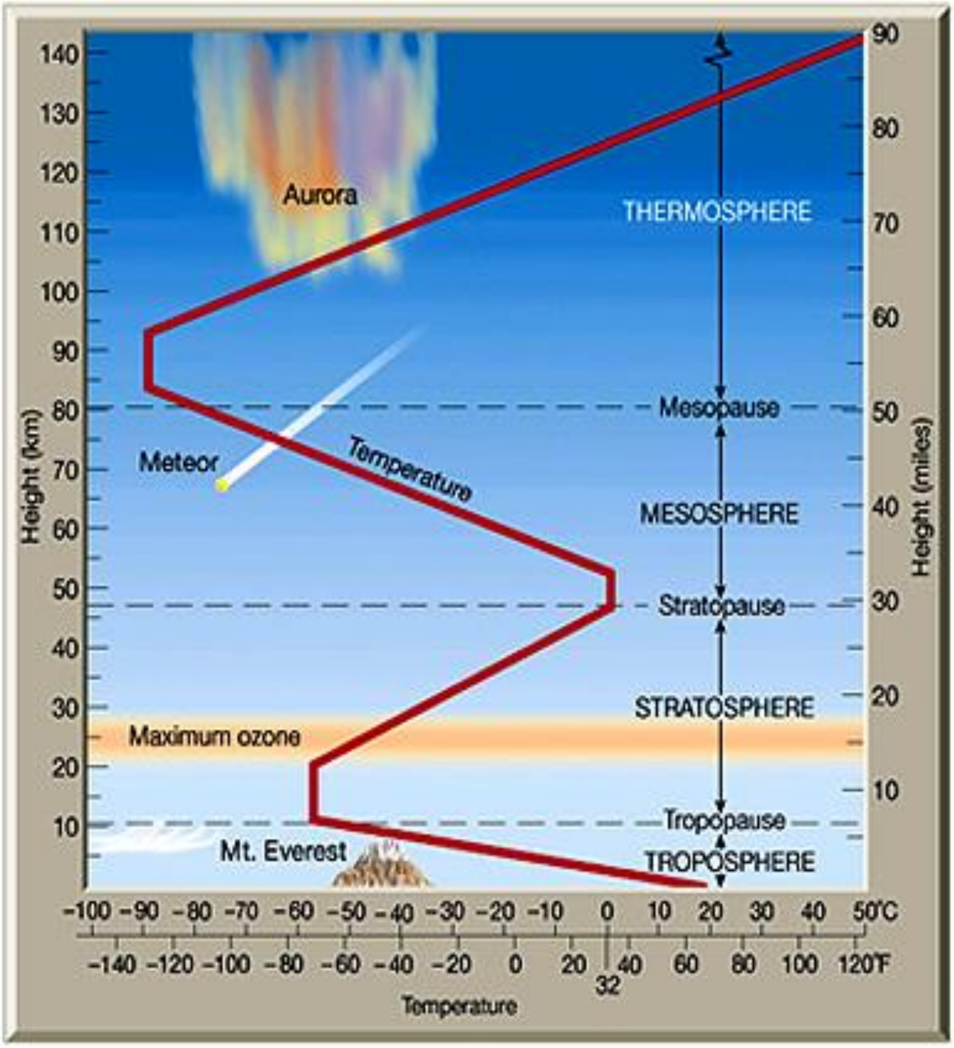
Thermosphere (85 km to 600 km):

- Extends up to 600km high.
- Temperature rises rapidly with height up to 300km.
- Due to sun's energy temperature rises with altitude

Exosphere (500 km to transition with interplanetary space, 10000 km):

- Begins at 500km and extends outward till inter plane space.
- Upper most layer of atmosphere.

- Merges into space in extremely this exosphere.



9. Explain about the NACA 2411 airfoil nomenclature with clear diagram. (Related)

<u>4-digit airfoils</u> (Ex: NACA ~~0~~2415)

NACA – National Advisory committee for Aeronautics.

1st digit (2) – indicates that the maximum camber is 0.02% over the chord.

$$\frac{\delta}{c} \cdot 100 = \% \text{ camber}$$

δ – camber of airfoil

c – chord of airfoil.

2nd digit (4) – The location of the maximum camber along the chord ling given as 0.4C.

last two digit (15) – the maximum thickness, here ~~0.15~~ 0.15C

$$\frac{t_{max}}{c} \cdot 100 = \% \text{ thickness}.$$

- For Example in this the symmetric Airfoil of 4 digit are should not to have camber then the first two digits are zero Ex:- NACA 0024, NACA 0010, etc..

10. Write a note on: Types drags

Types of Drags

Drag - The resistive force, acting in a direction opposite to the direction of motion.

The drag on an aircraft can be considered as the sum of many component drags. Drag can be divided into three major categories are

i) Parasite drag
- Profile drag
  - skin friction drag
  - pressure drag.
- interference drag

(ii) induced drag

(iii) wave drag

Parasite drag :- Parasite drag on an aircraft is the drag which is not caused by lift or compressibility effects.

Profile drag :- Measure of the resistance to flight caused by the air on the profile of the aircraft.

Skin friction Drag :- The skin friction drag is a drag caused by the viscosity of air flowing over the aircraft and is proportional to the shear stress of the air.

pressure drag :- The difference b/w the forces caused by the high pressures on the forward portion and low pressures on the aft portion of the aircraft is pressure drag.

Interference Drag :- Consider that the drag of a fuselage is measured in a wind tunnel and then the drag of a wing is measured in the same tunnel. When the fuselage and wing are mated and their total drag is measured in the wind tunnel, this drag would be greater than the sum of the two separately measured drags.

This difference is referred to as the interference drag and is caused by regions of turbulence and distracted airflow at the junctions of wing and fuselage.

Induced drag :- Induced drag is the aerodynamic drag induced or caused by the lift. For a broad understanding of induced drag, consider an aircraft in subsonic flight at an attitude to produce zero lift.

The total drag on the aircraft under these conditions is pure parasite drag. As soon as the aircraft produces lift, the drag increases and this increase is the induced drag.

This definition implies that the minimum drag on a subsonic aircraft in stabilised flight will occur when the aircraft produces zero lift. It also implies that for a given aircraft velocity the parasite drag is constant.

$$C_{Di} = \frac{C_L^2}{\pi e AR}$$

where e — Oswald's span efficiency

AR — Aspect ratio.

Wave Drag :-

wave drag, often referred to as compressibility or Mach drag, is the drag which results when flow over the surfaces of an aircraft exceeds Mach 1. Supersonic flow over wing and fuselage surfaces results in the formation of shock waves, causing a sizable increase in drag.

11. Define Speed of Sound and prove that $a = \sqrt{\gamma RT}$

The speed of sound is the distance travelled per unit time by a sound wave as it propagates through an elastic medium.

The speed of sound in an ideal gas depends only on its temperature and composition. The speed has a weak dependence on frequency and pressure in ordinary air, deviating slightly from ideal behavior.

Derivation of Speed of Sound Equation, $a = \sqrt{\gamma RT}$

By taking conservation of mass equation,

$$\dot{m} = \rho V A \quad \text{---(1)}$$

where, $\dot{m}$ = mass flow rate; ρ = density of air; A = flow area.

Similarly, the one dimensional conservation of momentum equation specifies,

$$-dp = \rho V dV \quad \text{---(2)}$$

where, dp = differential change in pressure
dV = differential change in velocity.

Let us assume that the flow area and mass flow rate are constant, and the particular velocity that we are going to determine is the speed of sound 'a', then,

$$\rho V = \rho a = (\rho + d\rho)(a + dV)$$

$$\rho a = (\rho a) + (\rho dV) + (a d\rho) + (d\rho dV) \quad \text{---(3)}$$

∴ The term in eqn (3), $d\rho dV$ is very small. so its neglected.

∴ the eqn (3) becomes,

$$0 = \rho dV + a d\rho$$

$$\rho dV = -a d\rho \quad \text{---(4)}$$

Substituting eqn (4) in (2),

$$\cancel{-}dp = \cancel{-}a d\rho . V \qquad |\because V = a.$$

$$\therefore dp = a^2 . d\rho \quad \text{---(5)}$$

for sound waves, the variations are small and nearly reversible. we can then evaluate the change in pressure from the isentropic relation,

$$\frac{dp}{p} = \gamma \frac{d\rho}{\rho} \quad \text{---(6)}.$$

where, γ = the ratio of specific heats. $= \frac{C_p}{C_v}$

Substituting eqn (6) in eqn (5).

$$\frac{\gamma . p . d\rho}{\rho} = a^2 d\rho \qquad \left| \because dp = \frac{p\gamma d\rho}{\rho} \right.$$

$$\frac{\gamma p}{\rho} = a^2 \quad \text{---(7)}.$$

from eqn of state,

$$p = \rho RT$$

or

$$\frac{p}{\rho} = RT \quad \text{---(8)}.$$

Substituting eqn (8) in (7).

$$\therefore \gamma RT = a^2$$

$$\Rightarrow \boxed{a = \sqrt{\gamma RT}}$$

12. Explain with neat sketches factor affecting Lift and Drag.

Factors affecting lift and drag

* A body that is pulled or pushed through the air causes a distribution in the air, and experiences a force.
* The amount of this force depends on the shape of the body.
* on its speed through the air.
* on its size
* on the smoothness of its surface.
* and on the density of the air through which it passes.
* That part of the force which is parallel to the direction of air flow is called drag.
* That part of the force which is at right angle to the direction in which the body is travelling is called lift.
* A wing is designed to give as much lift as possible with as little drag as possible.
* Defination of parasite drag. (write down)
* The motion produces a downwash which in turn causes the upward reaction or lift.
* The decrease in pressure on the top surface is caused by the speeding up of the flow over that surfaces.
* The center of pressure is therefore well forward.
* wings of high aspect ratio are the most efficient, because they have less induced drag.

13. *Problems*

Problem 14.25 *A jet plane which weighs 29430 N and has a wing area of 20 m² flies at a velocity of 250 km/hr. When the engine delivers 7357.5 kW. 65% of the power is used to overcome the drag resistance of the wing. Calculate the co-efficient of lift and co-efficient of drag for the wing. Take density of air equal to 1.21 kg/m³.* (A.M.I.E., Winter, 1981)

Solution. Given :

Weight of plane, $W = 29430$ N

Wing area, $A = 20\ \text{m}^2$

Velocity of plane, $U = 250\ \text{km/hr} = \dfrac{250 \times 1000}{60 \times 60}\ \text{m/s} = 69.44\ \text{m/s}$

Power delivered by engine $= 7357.5$ kW

Power required to overcome drag resistance

$= 65\%$ of $7357.5 = 0.65 \times 7357.5 = 4782.375$ kW.

Density of air, $\rho = 1.21\ \text{kg/m}^3$

Now, weight of plane $= \text{Lift force} = C_L \times A \times \dfrac{\rho U^2}{2}$

$$\therefore \quad 29430 = C_L \times 20 \times 1.21 \times \frac{69.44^2}{2}$$

$$\therefore \quad C_L = \frac{29430 \times 2}{20 \times 1.21 \times 69.44^2} = \mathbf{0.5046.\ Ans.}$$

Let F_D = Drag force

Power required to overcome drag resistance $= \dfrac{F_D \times U}{1000}$ kW

$$\therefore \quad 4782.375 = \frac{F_D \times 69.44}{1000}$$

$$\therefore \quad F_D = \frac{4782.375 \times 1000}{69.44} = 68870.6\ \text{N}$$

Now drag force, $F_D = C_D \times A \times \dfrac{\rho U^2}{2}$

$$\therefore \quad 68870.6 = C_D \times 20 \times \frac{1.21 \times 69.44^2}{2}$$

$$\therefore \quad C_D = \frac{68870.6 \times 2}{20 \times 1.21 \times 69.44^2} = \mathbf{1.18.\ Ans.}$$

14. Explain Lift Curve and Drag Curve.

Lift curve

Lift curve :-

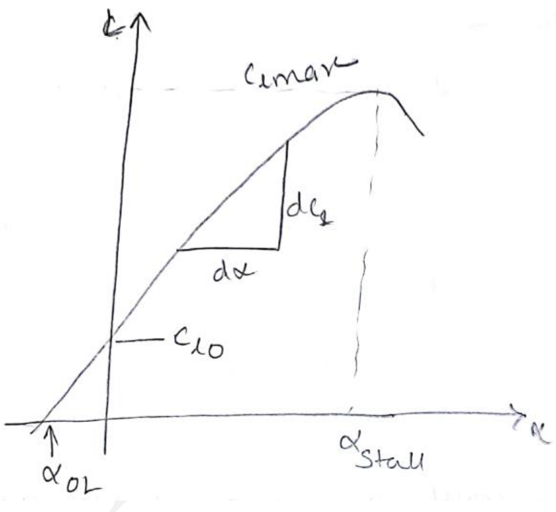
A plot of lift coefficient vs angle of attack is called the lift-curve.

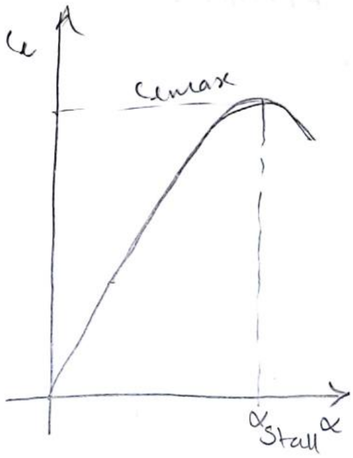
A typical lift curve as .

i) For small angles of attack, the lift curve is approximately a straight line. (almost linear assumptions)

2) For small angle of attack called the stall angle of attack, the lift coefficient reaches a maximum, $C_{l\,max}$.

3] There are two intercepts that we can designate, one the α axis for zero lift (α_{0L}), the zero-lift angle of attack, and the other zero angle of attack (C_{L0}), the lift at zero angle of attack.





$$a_0 = \frac{dC_l}{d\alpha}$$

Lift coefficient $C_l = C_{l0} + a_0\alpha = a_0(\alpha - \alpha_{0L})$

$\cancel{= a_0\alpha + a_0\alpha_{0L}}$

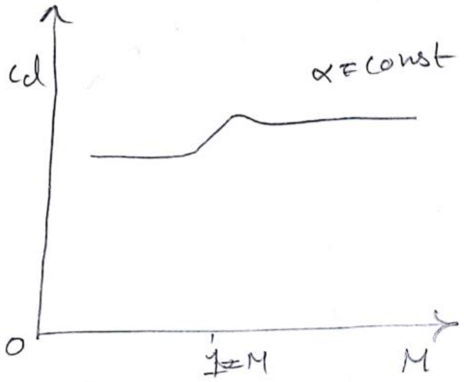
where $C_{l0} = -a_0\alpha_{0L}$

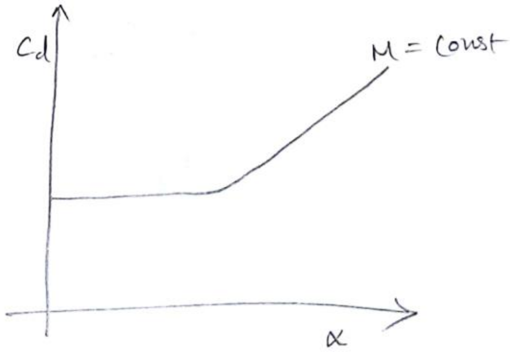
$$\alpha_{0L} = -\frac{C_{l0}}{a_0}.$$

Drag curve

The drag on an airfoil is primarily due to viscous effects at low speed and compressibility effects at high speed. At high angle of attacks, the flow can seperate from the upper surface and cause additional drag.

drag coefficient depends on Reynolds no, Mach no and angle of attack.





Drag coefficient is nearly const a subsonic speeds and tends to rise just before M = 1 (transonic region). Above M = 1.2 drag coefficient tends to increase or decrease slightly. The typical change of drag coefficient with angle of attack at a given Mach no, tends to increase slightly with angle of attack at low angles, and increases more rapidly at high angle of attack.

15. *Problems*

At 12 km in the standard atmosphere, the pressure, density and temperature are 1.9399×10^4 N/m^2, 3.1194 $\times 10^{-1}$ kg/m^3, and 216.66 K, respectively. Using these values, calculate the standard atmospheric values of pressure, density, and temperature at an altitude of 18 km, and check with the standard altitude tables.

Solution:

An examination of the standard temperature distribution through the atmosphere given in Fig. 3.3 of the text shows that both 12 km and 18 km are in the same isothermal region. Hence, the equations that apply are Eqs. (3.9) and (3.10) in the text. Since the two altitudes are in the same isothermal region we use the same base values of p and ρ, and the equations can be written as

$$\frac{\rho_2}{\rho_1} = \frac{p_2}{p_1} = e^{-(g_0/[RT])(h_2-h_1)}$$

where points 1 and 2 are any two arbitrary points in the isothermal region. Hence, with $g_0 = 9.8$ m/s^2 and $R = 287$ J/(kg K), and letting points 1 and 2 correspond to 12 km and 18 km, respectively, we have

$$\frac{\rho_2}{\rho_1} = \frac{p_2}{p_1} = e^{-(9.8/[287\times 216.66])(6000)} = 0.3884$$

Hence

$$p_2 = 0.3884 \times 1.9399 \times 10^4 = 7.53 \times 10^3 \text{ N/m}^2$$

$$\rho_2 = 0.3884 \times 3.1194 \times 10^{-1} = 0.121 \text{ kg/m}^3$$

and, of course,

$$T_2 = 216.66 \text{ K}$$

These answers agree with the table in Appendix A within round-off error.

16. *Problems*

At what value of the geometric altitude is the difference $h - h_G$ equal to 2% of the geopotential altitude, h?

Solution:

$|h - h_G|/h = 0.02 = |1 - h_G/h|$

Using Eq. (3.6), this relationship becomes

$|1 - (r + h_G)/r| = |1 - 1 - h_G/r| = 0.02$

Therefore

$h_G = 0.02r = 0.02 \times 6378 \times 10^3 = 127.56$ km

17. *Problems*

Given the pressure is P = 31,000 N/m^2, find the pressure altitude.

From the standard atmosphere tables:

@ h = 8700 m P = 32196 N/m^2 (lower)
@ h = 9000 m P = 30800 N/m^2 (upper)

Interpolate:

$$\frac{h - h_L}{h_u - h_L} = \frac{P - P_L}{P_u - P_L}$$

or

$$h = h_L + \frac{P - P_L}{P_u - P_L}(h_u - h_L)$$

$$h = 8700 + \frac{31{,}000 - 32{,}196}{30{,}800 - 32{,}196}(9000 - 8700)$$

$$= 8957 \text{ m}$$

Hence we are at a pressure altitude of 8957 meters.

18. *Problems*

7. Consider an airplane flying with a velocity of 60 m/s at a standard altitude of 3km. At a point on the wing, the airflow velocity is 70 m/s. Calculate the pressure at this point. Assume incompressible flow.

@ 3 km $P = 7.010 \times 10^4$ N/m^2 $\rho = 0.9090$ kg/m^3

Use Bernoulli's equation for incompressible flow:

$$P_1 + \frac{1}{2}\rho V_1^2 = P_2 + \frac{1}{2}\rho V_2^2$$

To get:

$$70100 + \frac{1}{2}(0.9090)(60)^2 = P_2 + \frac{1}{2}(0.9090)(70)^2$$

$$P_2 = 69509.2 \text{ N/m}^2$$