GATE Aerospace Coaching by Team IGC

AIRCRAFT STABILITY

"Any mechanical system when subjected to the small distance and if the system tends to maintain its original equilibrium state by applying restoring force, then the system is said to be stable".

If the system does not react to the disturbance and maintain its new position as an equilibrium state, then the system is said to be 'Neutrally stable'.

If the systems reacts to the disturbance and keep moving away from its original equilibrium state, then the system is called 'Unstable system'.

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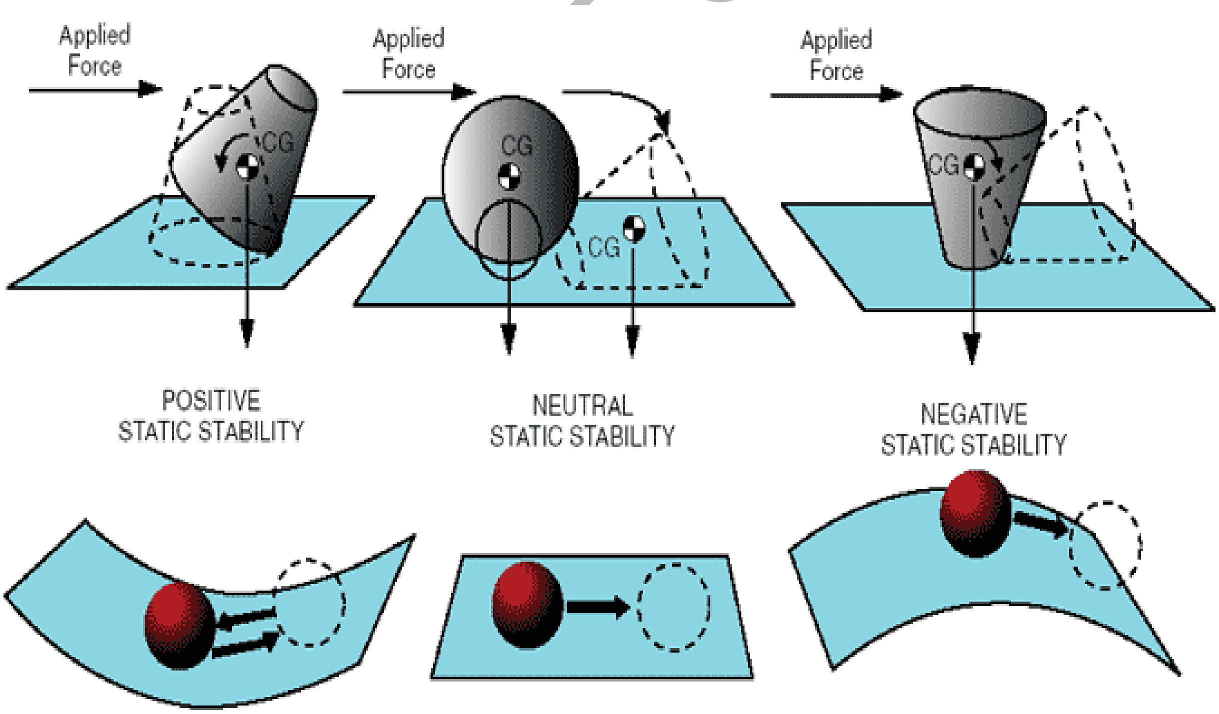
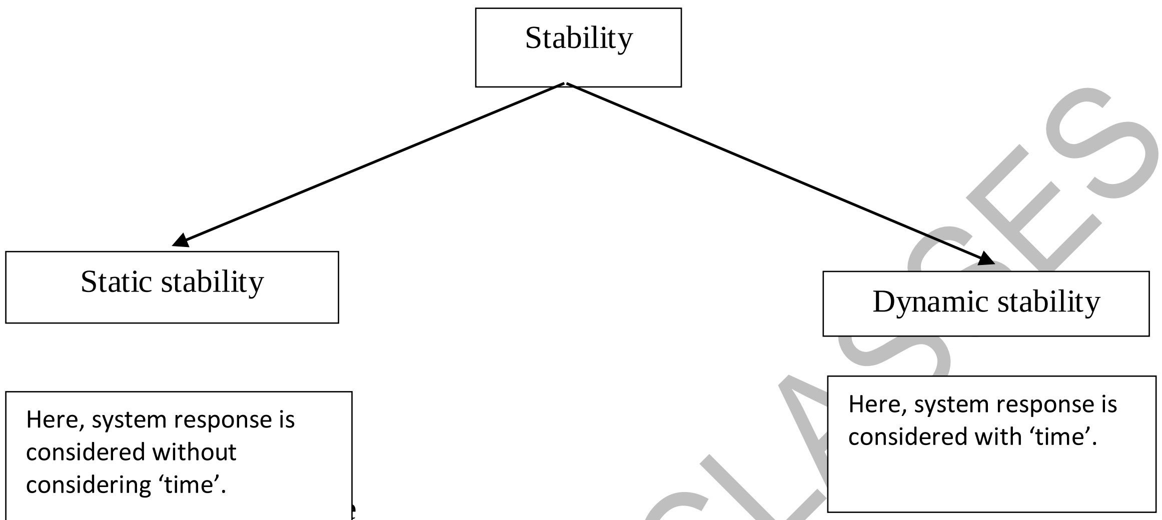
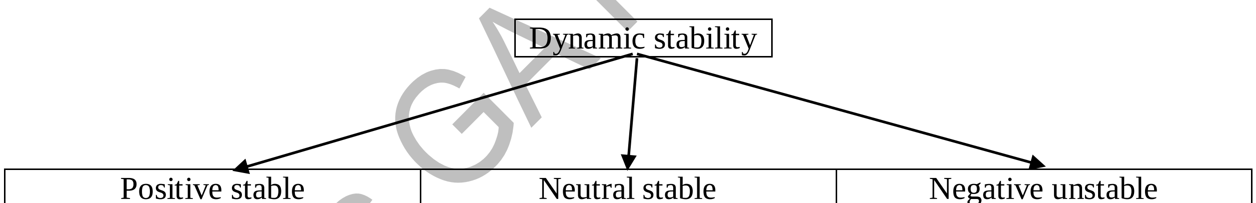




stability state without being dynamically stable that is a statically stable system may not be dynamically stable.



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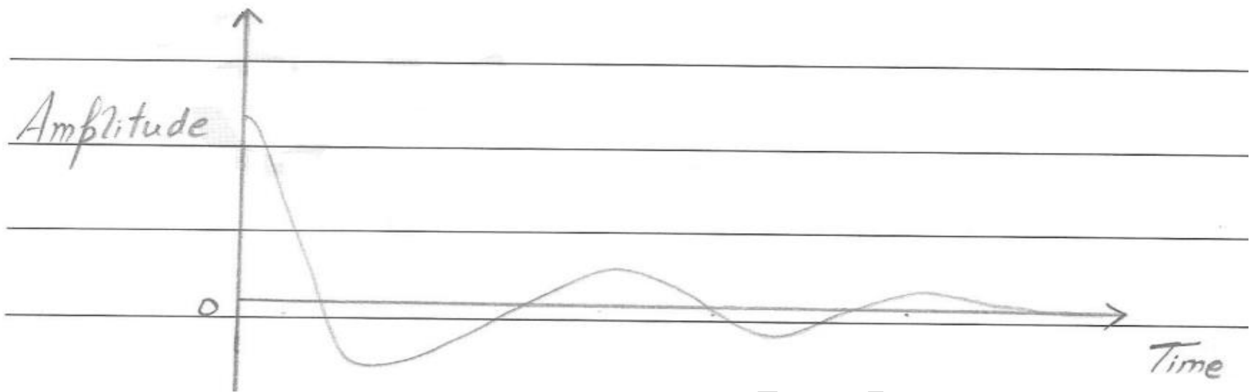
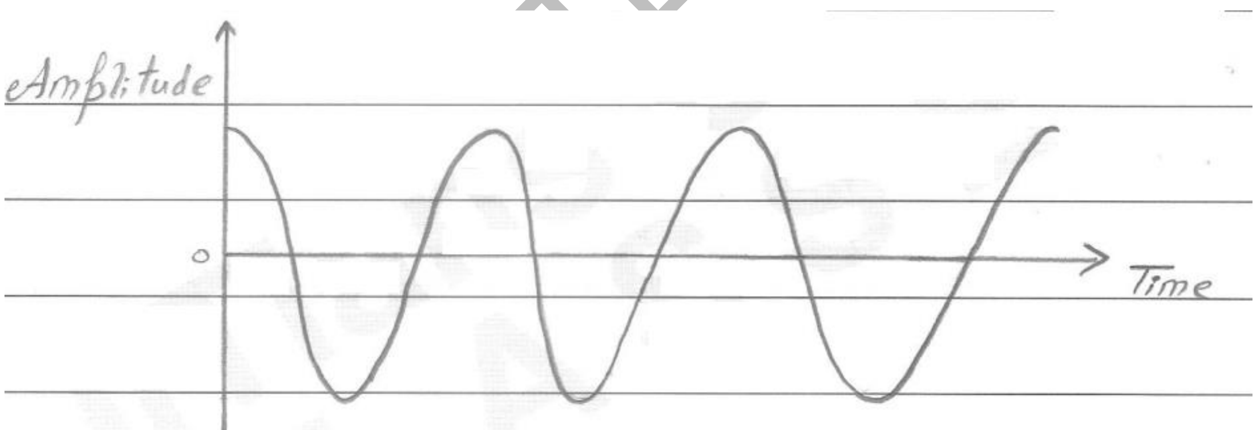
(A). Positive dynamic stability

When system is statically stable and tends to attain the equilibrium position by an "under damped" oscillation.



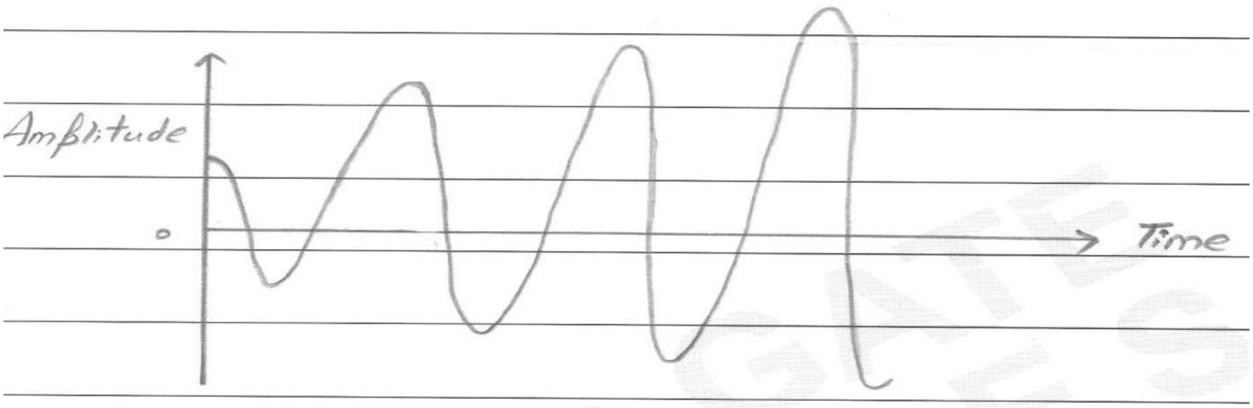
(B). Neutral Dynamic Stability

A mechanical system which tends to achieve original equilibrium position but keeps overshooting it by oscillating about it with a "constant amplitude".

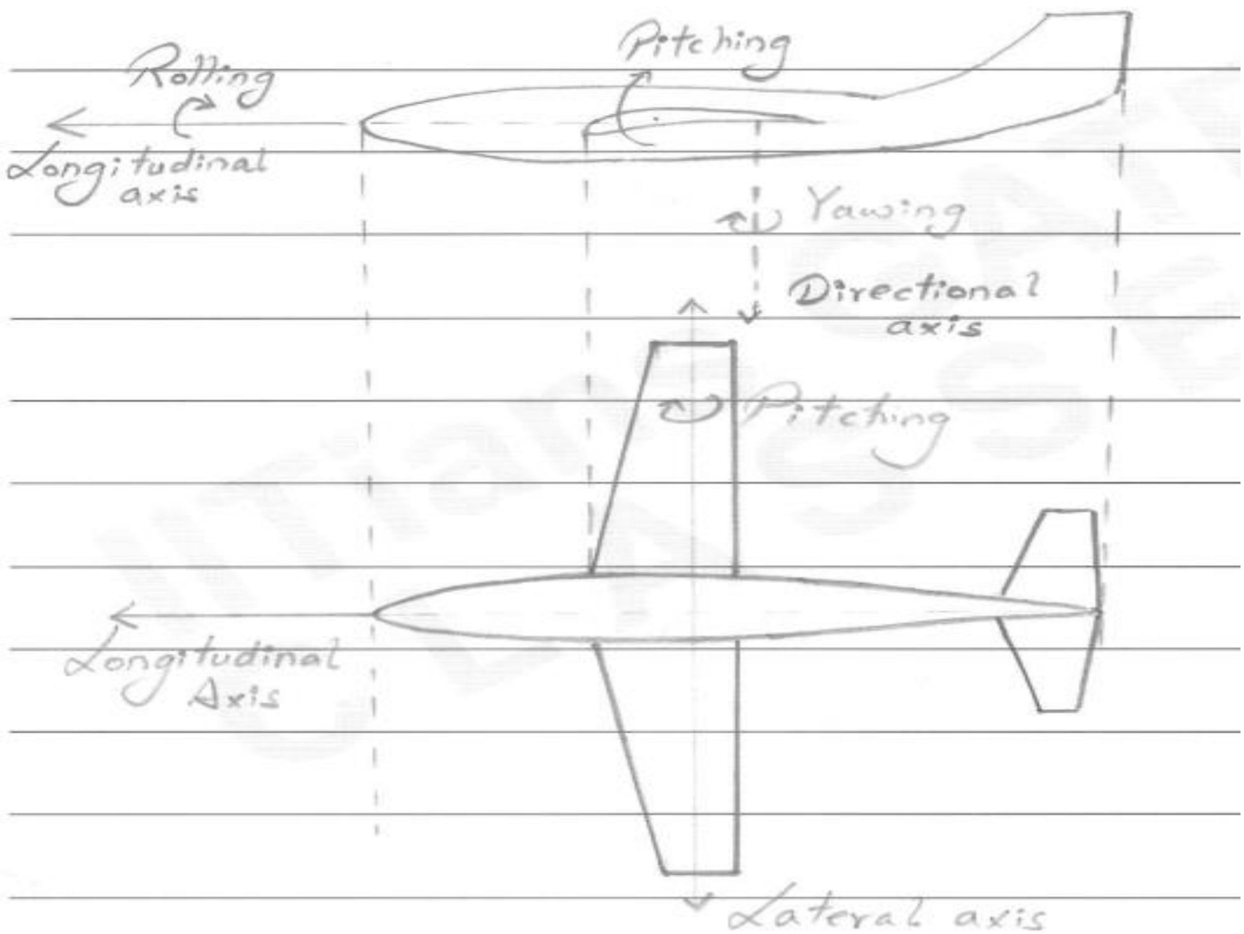


(C). Negative Dynamic Stability

A mechanical system which tends to achieve original equilibrium position but keep overshooting it by oscillating about it with "increasing amplitude with time".



Parameter Notation



Sign notation

With respect to C.G of an aircraft

For pitching,

Nose up-positive sign (+ve)

Nose down-negative sign (-ve)

For rolling,

Right wing (star board side) down (+ve)

Right wing (star board side) up (-ve)

For yawing,

Turning towards right wing (+ve)

Away from right wing (-ve)

Moments on the Airplane:-

A study of stability and control is focused on moment

The moment can be taken about any ability point,(i.e) leading edge, taken edge or Aerodynamic center of the wings.

Moment at aerodynamic center:-

As per definition of aerodynamic center, the pitching moment coefficient remains constant with all angle of attack.

$$C_{mac} = \frac{\mu_{ac}}{q_\infty s_c}$$

Now at zero lift coefficient for a flying wing

$$C_{mac} = (C_{m,c/a})_{L=0} = C_{m,anypoint_{L=0}}$$

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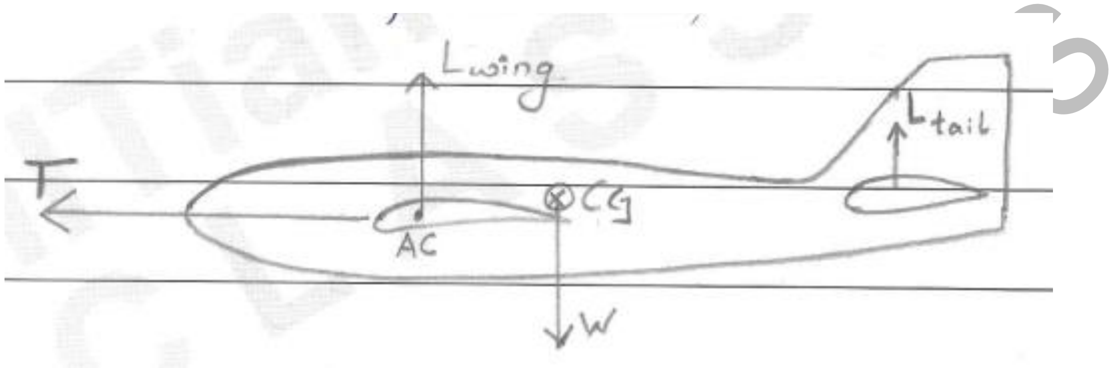
Hence C_{mac} can be obtained from the value of the moment coefficient about any point when the wing is at the zero lift angle of attack. $(\alpha_{L=0})$

Moment on airplane:-



Consider the system of force as shown in figure

Moment about C.G an aircraft will be calculated for L, D and μ_{ac} of the wing, thrust of the aircraft, lift of the tail and aerodynamic forces and moments on the parts of the aircraft.

$$M_{cg} = f(L, D, T, L_{Tail}, \text{other forces})$$

Here, weight of the aircraft will not be considered for obvious reasons

$$C_{mcg} = \frac{M_{cg}}{q_\infty s_c} = \frac{f(L, D, T, L_{Tail}, other\, forces)}{q_\infty s_c}$$

All the discussions of the stability of an airplane will be done by considering C_{mcg}.

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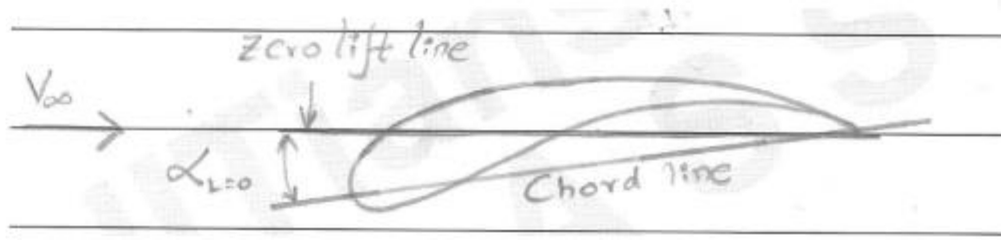
Absolute angle of attack :-

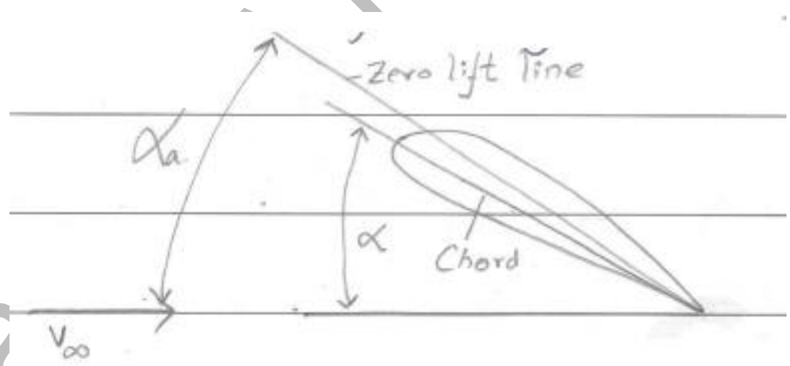
Zero lift line :-

Consider a wing at an angle of attack such that lift is zero, that is the wing is at the zero lift angle of attack $(\alpha_{L=0})$, as shown in figure.



A line drawn through the trailing edge parallel to the relative wind U_{∞} is called "zero lift line".

Now if wing is pitched to some angle of attack, when there is a positive lift.



Where,

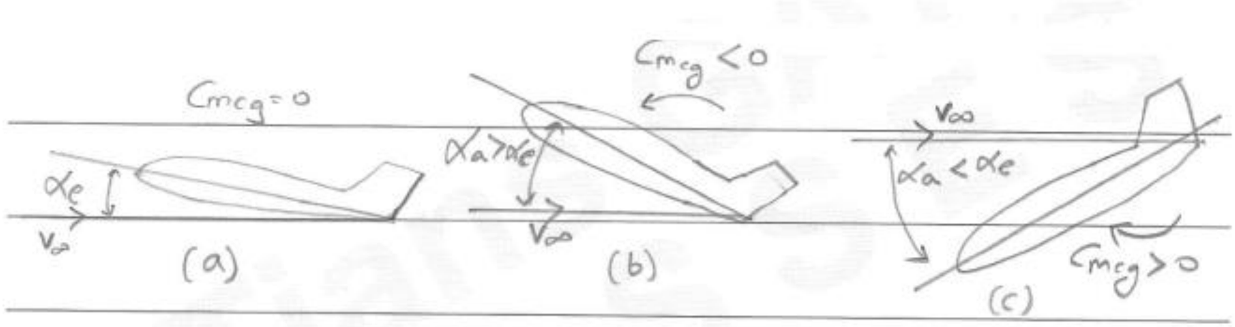
$\alpha_a \Rightarrow$ Angle between zero lift line and free stream velocity.

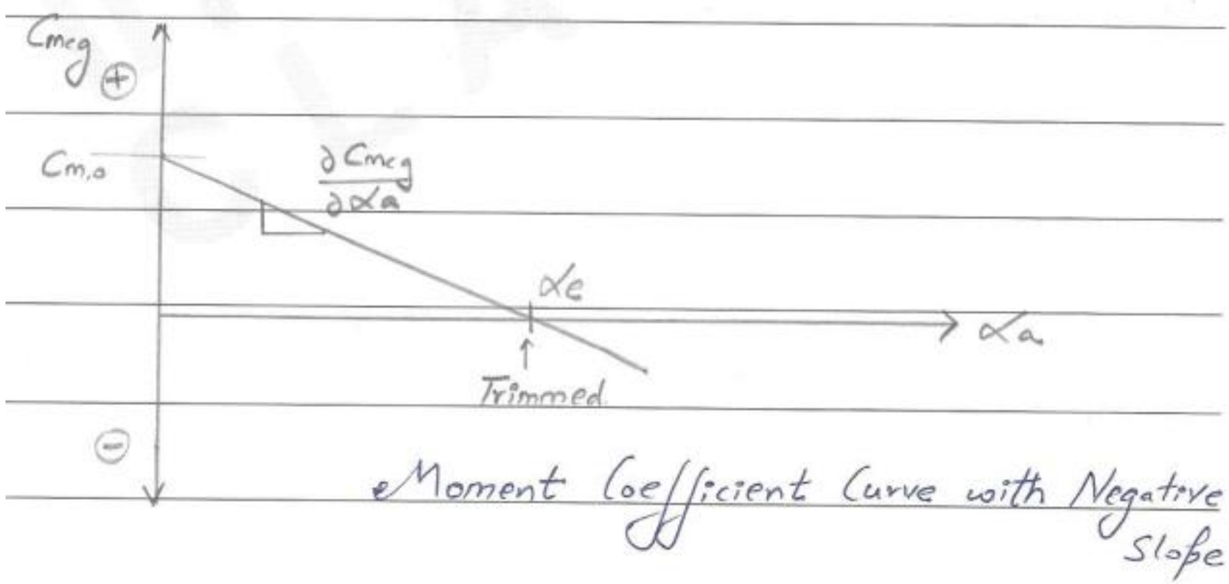
$\alpha_a = \alpha + \alpha_{L=0} \Rightarrow$ Absolute Angle of attack

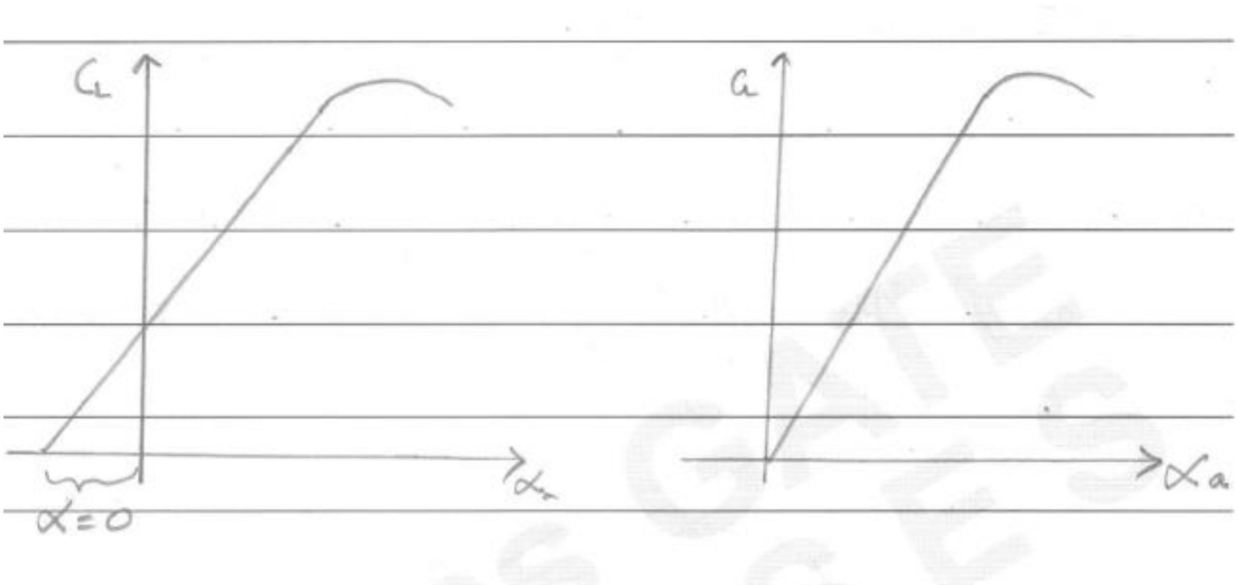
"The angle between zero lift line and free stream velocity is called Absolute angle of attack".

Criteria for longitudinal static stability :-

Consider a rigid airplane with fixed controls, is being tested in a wind tunnel or free flight, then its variation of M_{cg} with angle of attack has been measured.







When,

$\alpha_a = 0 \Rightarrow L = 0$ Irrespective of camber

Statically Stable :-

If an aircraft tends to move back towards its equilibrium position after being disturbed, then it is called statically stable.

If aircraft is nose up, it should have negative moment and vice versa.

$\frac{dC_{m_{cg}}}{d\alpha} < 0$ for static stable

if $\frac{dC_{m_{cg}}}{d\alpha} = 0$ Statically Neutral

if $\frac{dC_{m_{cg}}}{d\alpha} > 0$ Statically Unstable

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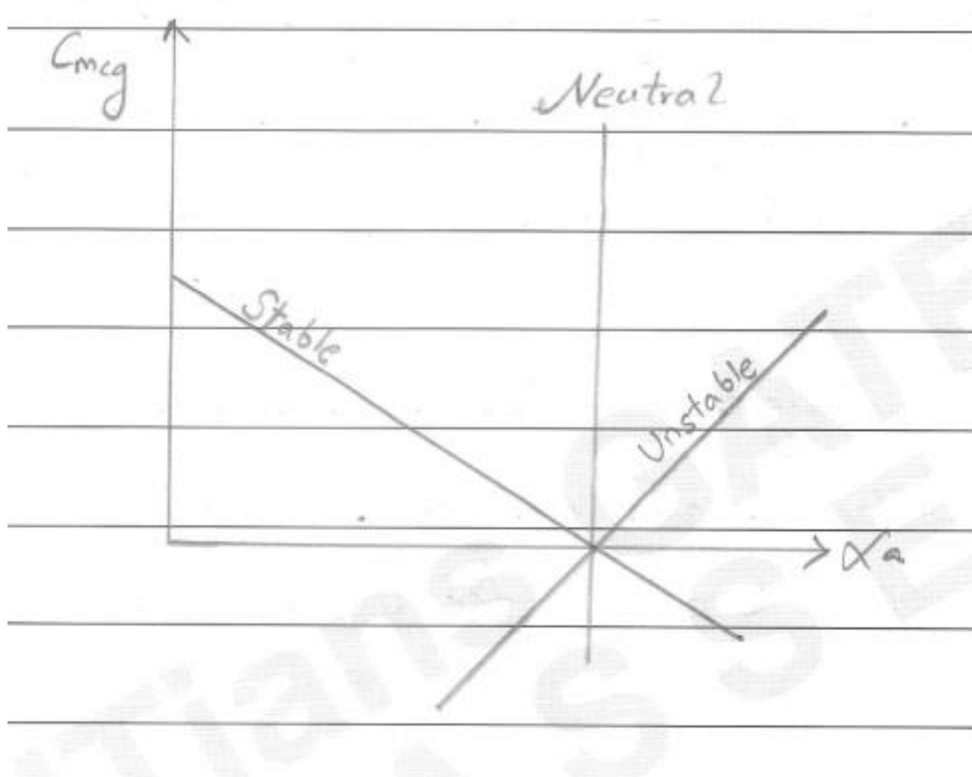
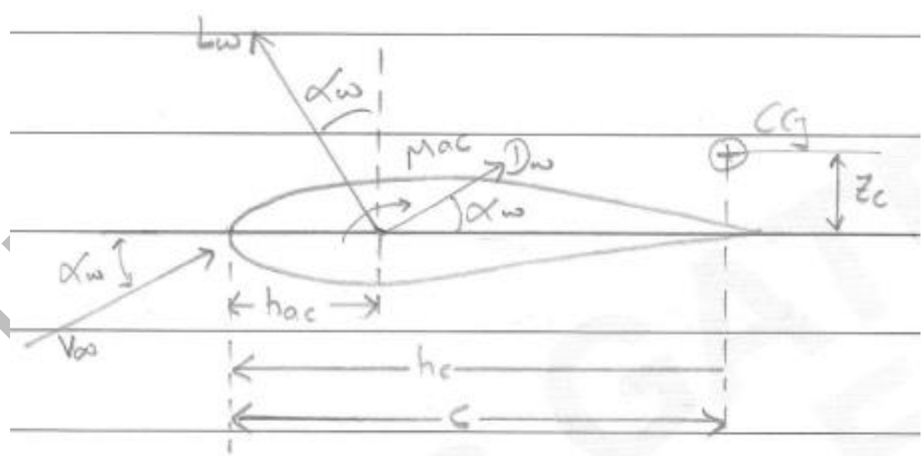


Contribution of the wing M_{cg} :-



Taking moment about C.G

$$M_{cg} = M_{ac_w} + \alpha_w cos\alpha_w (h_c - h_{ac}) + \alpha_w sin\alpha_w \xi_c + D_w \sin\alpha_w (h_c - h_{ac}) - D_w cos\alpha_w \xi_c$$

For normal flight operation α_w is small

$$cos\alpha_w = 1 \quad ; \quad sin\alpha_w \cong \alpha_w$$

$$M_{cg} = M_{ac} + (L_w + D_w\alpha_w)(h - h_{ac_w}) + (L_w\alpha_w - D_w)\xi_c \rightarrow (2)$$

h and z are coordinate of C.G in the function of chord length

Dividing equation (2) by '$q_\infty Sc$'

$$C_{m_{cg}} = C_{m_{ac}} + (C_{L_w} + C_{D_w}\alpha_w)(h - h_{ac_w}) + (C_{L_w}\alpha_w - C_{D_w})\xi \rightarrow (3)$$

Now ξ is very small and can be neglected. Also $C_{D_w}\alpha_w$ is very small number

$$C_{m_{cg}} = C_{m_{ac}} + C_{L_w}(h - h_{ac_w}) \rightarrow (4)$$

Now

$$C_{L_w} = \left(\frac{dC_{L_w}}{d\alpha}\right)\alpha_w = a_w\alpha_w$$

a_w = lift slop of the wing

$$C_{m,cg} = C_{m_{ac_w}} + a_w\alpha_w(h - h_{ac_w}) \rightarrow (5)$$

$$C_{m,cg} = f(C_{m_{ac}}, a_w, \alpha_w, h, h_{ac_w})$$

If fuselage is considered with wins

Wing body configuration

$$C_{L_{wB}}, \alpha_{wB}, a_{wB}, h_{ac_{wB}}$$

$$C_{m,cg,wB} = C_{m_{ac_{wB}}} + a_{wB}\alpha_{wB}(h - hac_{aw})$$

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Problems

Qus). A wing body model is tested in a subsonic wind tunnel. The lift is found to be zero at a geometric angle of attack $\alpha = -1.5^o$. at $= 5^o$, the lift coefficient is measured as 0.52.

$\alpha = 1.0^o$ moment coefficient $M_{cg} = -0.01$

$\alpha = 7.88^o$ $M_{cg} = 0.05$

The center of gravity is located at 0.35 .calculate the location of aerodynamic center and the value of $C_{m,ac,wB}$

Sol :-

Lift slope $a_{wB} = \frac{dC_L}{d\alpha} = \frac{0.52-0}{5-(-1.5)} = 0.08\, per\ degree$

$$C_{m_{cg_{wB}}} = C_{m_{ac_{wB}}} + a_{wB}\alpha_{wB}\left(h - h_{ac_{wB}}\right) \to (1)$$

At $\alpha = 1.0^o$ here α is the geometric angle of attack

α_{wB} is the Absolute angle of attack

$\alpha_{wB} = \alpha + \alpha_{L=0}$

$= 1.0^o + 1.5^o = 2.5^o$

Using (i)

$C_{m_{cg}} = -0.01$ for $\alpha = 1.0^o$

$$-0.01 = C_{m_{ac_{wB}}} + 0.08(2.5)\left(h - h_{ac_{wB}}\right) \to (2)$$

At $\alpha = 7.88^o$ $C_{m_{cg}} = 0.05$

$$0.05 = C_{m_{ac_{wB}}} + 0.03(7.88 + 1.5)\left(h - h_{ac_{wB}}\right) \to (3)$$

Solving equation (2) and (3) for $C_{m_{ac_{wB}}}$ and $h_{ac_{wB}}$ we have $h = 0.35$

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$$h_{ac_{wB}} = 0.29$$

Also,

$$C_{m_{ac_{wB}}} = -0.032$$

Aerodynamic center is located at $0.29c$

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