Subsonic Aerodynamics
Basics, Laws and Definitions

Bernoulli's equation:

Total Pressure Constant = **Dynamic Pressure** ($\frac{1}{2} \rho V^2$) **+ Static Pressure**

The law of **conservation of mass** states that for any **system closed**, the mass of the system must remain constant over time. The **mass can neither be created nor destroyed**, although it may be rearranged in space.

Bernoulli's theorem states that:
In a streamlined flow of fluid, the sum of all energies is a constant
The sum of static and dynamic pressure is constant.
Static pressure plus dynamic pressure is constant.
The sum of pressure energy and dynamic energy is constant.

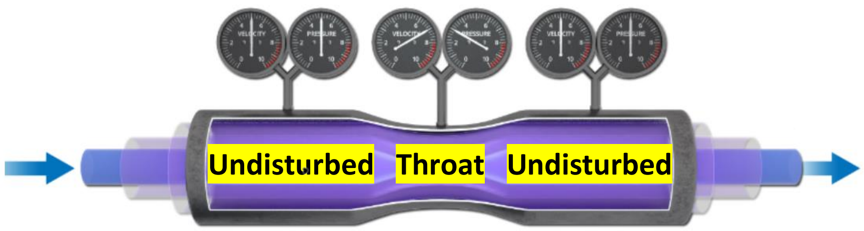
Dynamic Pressure is egual to **($\frac{1}{2} \rho V^2$)** **$q = (\frac{1}{2} \rho V^2)$**
Bernoulli's equation can be written as **PT = PS + q**
Bernoulli's equation can be written as **PT - q = PS**
Bernoulli's equation can be written as **PSTAT + ½rho × TAS² = constant**

The dynamic pressure is zero when velocity is zero.
The dynamic pressure increases as static pressure decreases.

If temperature of a **gas** is kept constant and **pressure increases**, the **density increase.**
If **density** is kept **constant**, the **dynamic** pressure **increases** proportionally with the **square of the velocity. ($\frac{1}{2} \rho V^2$)**

Given:
p = pressure
rho = density
T = absolute temperature
p / (rho * T) = constant

Considering subsonic **incompressible airflow** through a Venturi



The static pressure in the undisturbed airflow is higher than in the throat.
The static pressure in the throat is lower than in the undisturbed airflow
The dynamic pressure in the throat is higher than in the undisturbed airflow

The dynamic pressure in the undisturbed airflow is lower than in the throat.
The speed in the undisturbed airflow is lower than in the throat.
The speed of the airflow in the throat is higher than in the undisturbed airflow.
The total pressure in the undisturbed airflow and in the throat is the same.

In the throat of a venturi the static pressure is lower and the speed is higher compared to the undisturbed airflow.
In the throat of a venturi the dynamic pressure is higher and the total pressure is the same compared to the undisturbed airflow.
In the undisturbed airflow the dynamic pressure is lower and the total pressure is the same compared to the throat.
In the undisturbed airflow the static pressure is higher and the speed is lower compared to the throat.

In a **convergent tube** with an incompressible sub-sonic airflow,
PS decreases, PDYN increases, PTOT remains constant.
The (subsonic) **static pressure decreases in a flow in a tube when the diameter decreases.**
If the **cross-sectional** area of a tube **decreases**, the **speed** of the flow **increase**
If the **cross-sectional** area of a tube **increases**, the **speed** of the flow **decrease**

A subsonic **incompressible** flow, as air flows through a **tube of increasing cross-sectional** area the air **density does not vary. (only pressure vary)**
If the continuity equation is applicable: if the cross sectional area of a tube changes (low speed, subsonic and incompressible flow) the air density (rho) is **rho1 = rho2. (density entering is egual to density leaving a venturi, density is constant)**

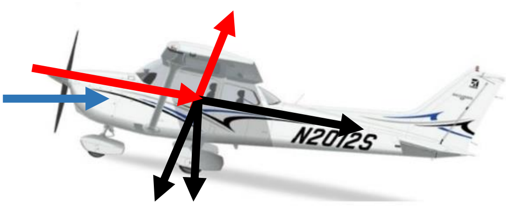
Density of the atmosphere will **decrease with increase in humidity. (more humidity, less air)**

An **increase** in the flow's **temperature** will **decrease the mass flow.**

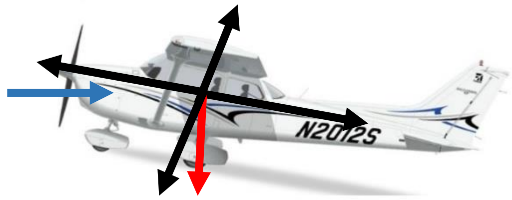
Static pressure acts **in all directions.**

The **lift and drag** forces, acting on an Aerofoil cross section **depend on the pressure distribution around the aerofoil cross section.**

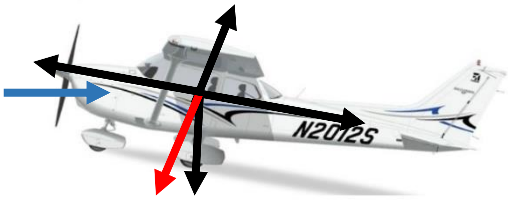
In a fluid disregard horizontal plane



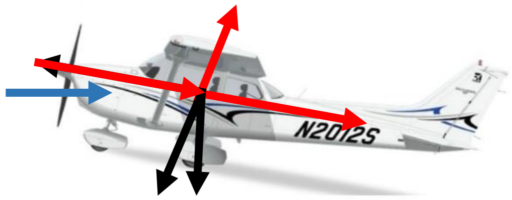
(Lift always perpendicular to undisturbed airflow)
Lift is the component of **the total aerodynamic force, perpendicular to the undisturbed airflow.**
Lift is the force that act at **right angle** to **relative airflow**



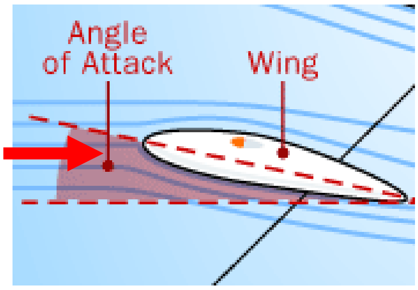
Weight is a **force**. (weight = m x g10) (9.81)



Wing **loading** is the ratio of **aircraft weight to wing area.**



Drag is acting in the direction of relative wind (airflow) and lift is perpendicular to the relative wind (airflow).



The **angle of attack** is the
angle between wing chord line and direction of the relative airflow.
angle between chord line and the relative undisturbed airflow.
angle between chord line and the undisturbed airflow
angle between chord line and the free stream

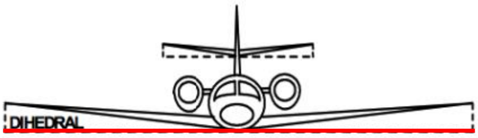
The **aeroplane angle of attack** is the **angle between wing root chord line and longitudinal axis**

An **aeroplane's angle of incidence** is the **angle between its speed vector and longitudinal axis.**

Relationship between force (F), acceleration (a) and mass (m) is **F = m × a**
Weight is a Force (mass x g acceleration)

The weight of an object depends on the acceleration due to gravity.
The mass of an object is independent of the acceleration due to gravity.
The weight of a body can be determined by multiplying its mass by the acceleration due to gravity. F = m × a
The mass of a body can be determined by dividing its weight by the acceleration due to gravity.

The units of wing loading are (W/S) **N/m^2** and dynamic pressure (q) **N/m^2**

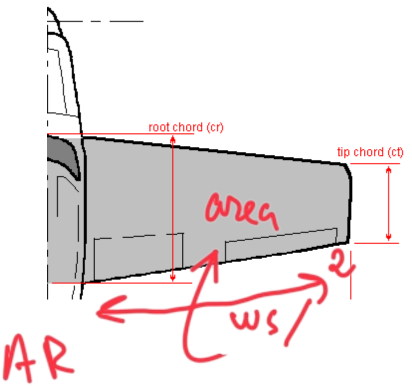


Dihedral of a wing is the **angle** between **the wing plane and the horizontal with the aeroplane in an unbanked, level condition.**
Dihedral of the wing is the angle between **the 0.25 chord line of the wing and the lateral axis.**

The construction feature of a wing called "**wash out" (wing twist**) is **a decrease in the angle of incidence from root to tip.**
Geometric washout (wing twist) means that **the tip of the wing has lower angle of attack than the root.**

The wings are normally with a 4° angle of incidence, but angle is different from tip to root by twisting the wing down near the root.

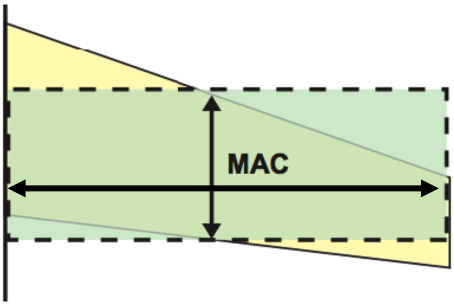
Taper Ratio = Tip / Root



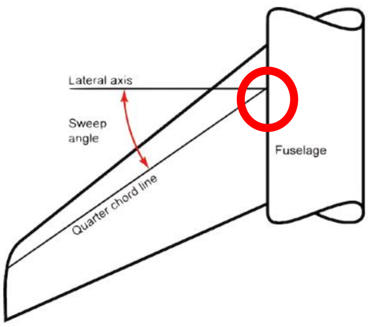
A wing is said to be **tapered** if **the chord at the wing tip is less than the chord at the root.**

Taper ratio is the ratio of **the tip chord to the root chord.**

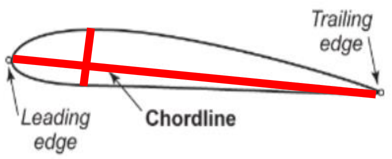
Aspect Ratio = wingspan ÷ mean chord
Aspect Ratio = (wingspan)^2 ÷ wing area



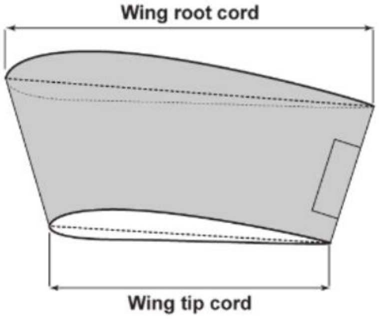
The **aspect ratio** of the wing is the ratio between **wingspan and mean geometric chord (MAC).**
Aspect ratio of a wing is the ratio between **wingspan squared and wing area.**
The **Mean geometric chord is the wing area divided by the wingspan.**
The **Mean Aerodynamic/geometric Chord (MAC)** is the **chord of an equivalent untwisted, rectangular wing with the same pitching moment and lift characteristics as the actual wing.**



Wing **sweep angle** is the angle between **the quarter-chord line of the wing and the lateral axis.** **(misured at ¼ cord)**
A wing would be said to be **swept** back if **the quarter chord line was inclined backwards from the lateral axis.**

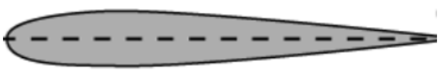


The **thickness to chord ratio** of an aerofoil **is the ratio of the maximum thickness of an aerofoil section to its chord.**
thickness ÷ chord

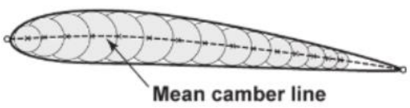


The relative **thickness** of an aerofoil is expressed in **% chord.**

The **chord** line of an aerofoil section is **the line, drawn from the leading edge to the trailing edge.**



In a **symmetrical** Aerofoil the **mean camber** line is **a straight line co-incident with the chord line.**

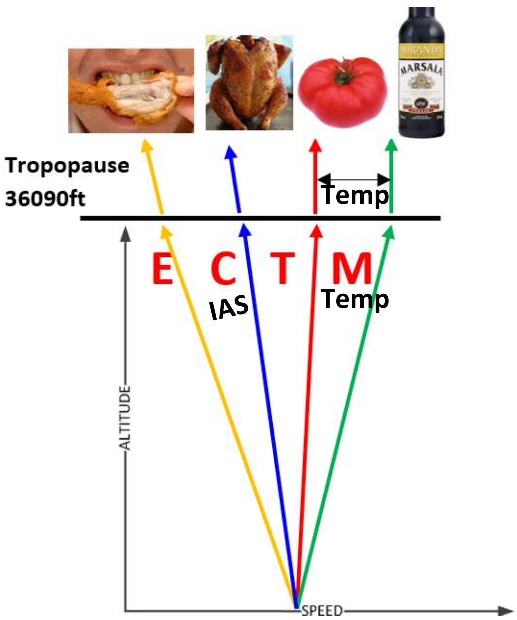


An aerofoil is **cambered** when **the line, which connects the centres of all inscribed circles, is curved.**
Camber of an aerofoil section is **the largest distance between the chord line and the camber line.**

The **mean camber line** of an aerofoil section **is a line from the leading to the trailing edge equidistant from the upper and lower surfaces.**
The camber line is a line connecting the **leading** and **trailing** edge **midway** between the upper and lower surface of a aerofoil

Indicated airspeed (IAS) = the direct instrument reading obtained from the ASI
Calibrated airspeed (CAS) = IAS corrected for position (pressure) and instrument errors
Equivalent airspeed (EAS) = the speed at sea level, is CAS corrected for compressibility error
True airspeed (TAS) = EAS corrected from Density error

Summary Errors correction:
IAS
CAS = **Instrument | Position**
EAS = **CAS | Compressibility**
TAS = **EAS | Density**
IP **CD**



ikka
Eat Chicken Tomato Ma~~r~~sala

The **true airspeed (TAS)** is **lower than the indicated airspeed (IAS) at altitudes below sea level, under ISA conditions. (in red sea there is a depression)**

At a constant TAS the dynamic pressure **will be greater at sea level than at high altitude.**

Two-dimensional Airflow Around an Aerofoil

Lift theory Fluid dynamic: The lift force, acting on an aerofoil is mainly caused by reduced pressure **on the upper side of the aerofoil.** (suction)

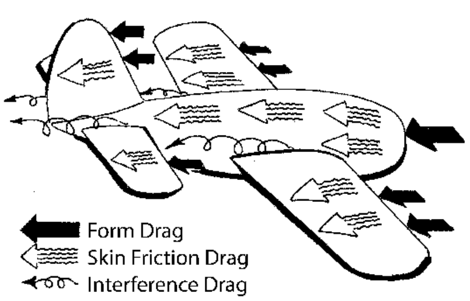
Lift theory Mechanical: The Newton 3 law principal of action and reaction say that lift in created from an Aerofoil with an angle of attack. The air is deflected downward by the action (angle) of the Aerofoil, and in reaction the wing is pushed upward.

Lift is generated when:
the flow direction of a certain mass of air is changed.

Of the **total lift** produced by the wing **the upper surface produces the greater proportion at all speeds.**

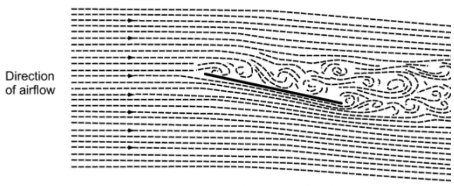
Typically the **C_L/C_D ratio** is **maximum at an angle of attack of 4°.**
(C_L/C_D ratio = Lift/Drag   at AoA of 4° there is max lift)

The **profile drag** of a wing section or blade element is **proportional** to:
the square of the relative velocity of the air. V^2
the drag coefficient. C_D
the air density. ½ ρ



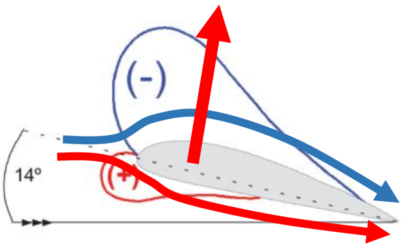
The total drag of an aerofoil in **two dimensional** flow comprises **form/profile/Pressure drag and skin friction drag.**

The total drag of an aerofoil in **three dimensional** flow comprises **form/profile/Pressure drag, skin friction drag and induced drag**
Induced drag, vortices in three dimensional flow



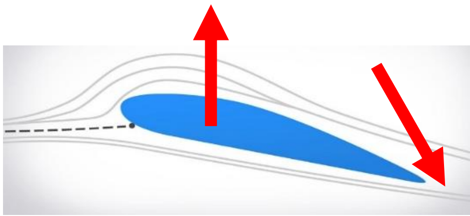
Airflow over a flat plate

A **flat plate**, when positioned in the airflow at a small angle of attack, will produce **both lift and drag.**
Every object in airflow produce some lift and drag



As the air flows over the upper surface of an aerofoil **its** speed increases, static **pressure** decreases, total pressure is constant.

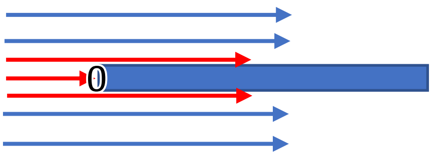
In an aerofoil with a positive camber and a positive angle of attack the highest flow velocities occur in the upper side



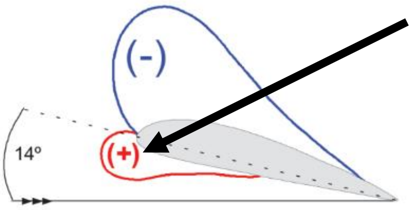
With increasing downwash, lift generated by the aerofoil increases.

An aerofoil which is producing lift will have **upwash ahead of the wing and downwash behind it.**

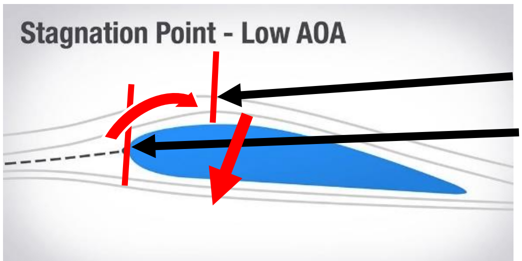
Stagnation point is where flow is **separated** by the presence of an object and speed is **zero**.
Stagnation point is **the point where the velocity of the relative airflow is reduced to zero.**



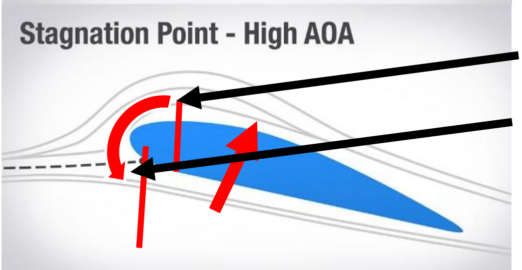
Consider a **flat plate** with **zero angle** of attack. The boundary layer is the region in which **the velocity varies between the speed of the undisturbed airflow and zero.**



At the **stagnation point** the **static pressure reaches a maximum value.**

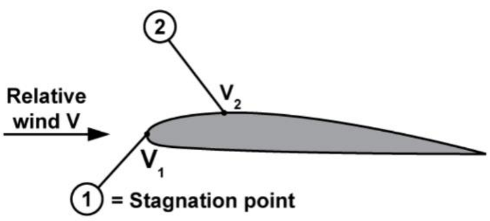


as the **angle of attack decreases**
The point of lowest static pressure moves aft.
The stagnation point moves up. Moves forward

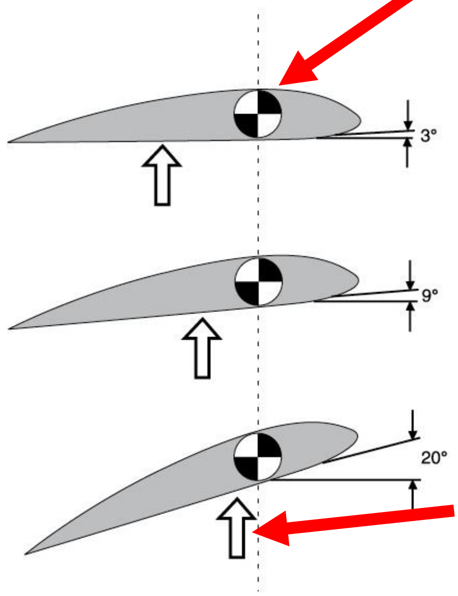


as the **angle of attack increases**
The point of lowest static pressure moves forward.
The stagnation point moves down. Moves backward
As the angle of attack increases, the stagnation point on the aerofoil's profile moves downwards on the profiles leading edge.

Static pressure and stagnation point move along the surface of the wing according to AoA

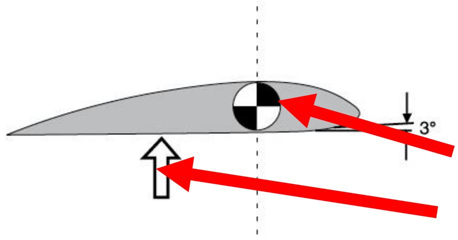


The **speed** at stagnation point is zero and in upper part is faster:
V1 = 0 and V2 > V

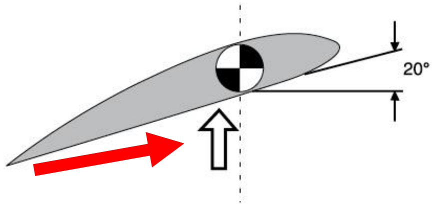


The aerodynamic centre is the point which **the pitching moment coefficient does not change with varying angle of attack.**
The location of the aerodynamic centre of an aerofoil section **is at approximately 25% chord irrespective of angle of attack.**

The point, where the single resultant **aerodynamic** force acts on an aerofoil, is called **centre of pressure.**
The point, where the **aerodynamic** lift **acts** on a wing is **the centre of pressure.**
The **Center of Pressure** of an aerofoil section **is the point where the resultant aerodynamic force is applied.**
The **Center of Pressure** is the point on the **chord line** through which the resultant of **all aerodynamic forces acts**.

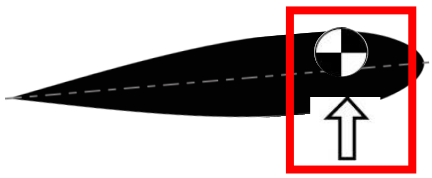


When **angle of attack is decreased** the **centre of pressure** will **move backward**
When **speed is increased** in straight and **level flight** (lift constant)
the magnitude of the total lift force remain the same
centre of pressure move aft

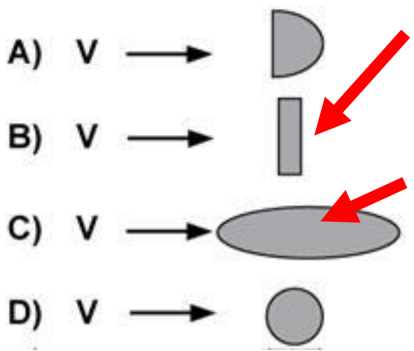


When **angle of attack is increased** the **centre of pressure** will **move forward**
When **speed is decreased** in straight and **level flight** (lift constant)
centre of pressure move forward
the magnitude of the total lift force remain constant

The location of the **centre of pressure** of a positively cambered aerofoil section at **increasing angle** of attack will **shift forward until approaching the critical angle of attack.**



In a **symmetric Aerofoil center of pressure & aerodynamic centre are at approximately 25% chord irrespective of angle of attack. They doesn't change with angle of attack**



Body B body will have the **highest pressure drag/form drag**

Body C body will have the **lowest pressure drag/form drag**
(the wind come from left, B have more pressure drag because the back is not rounded and create more drag)

Order of decreasing drag is B, A, D, C
Order of increasing drag is C, D, A, B

If in a flow the **streamlines converge**, the **static** pressure in the flow will **decrease** and **velocity will increase**
The static pressure decreases as the streamlines converge.
The velocity increases as the streamlines converge.

Ice is most likely to form on the leading edge of the wing where the stagnation point is located.

Coefficients

V^2=TAS^2 |q d pressure| Area | Chamber/AoA

$$\text{LIFT} = (\tfrac{1}{2}\rho\, V^2) \quad S \quad \underline{C_L}$$

$$\text{DRAG} = (\tfrac{1}{2}\rho\, V^2) \quad S \quad \underline{C_D}$$

The terms **q** and **S** in the lift formula are **dynamic pressure and the area of the wing.**
C_L varies with **angle of attack**

The correct drag formula is **DRAG = C_D × ½ RHO × V^2 × S**

If the lift generated by a given wing is 1 000 kN, **if the wing area is doubled lift is doubled** 2000 kN

Lift is a function of **velocity, wing area, C_L and density.**

At a given TAS, **increase of air density Lift and drag will increase.**

For a given angle of attack the **lift/drag ratio is unaffected by density changes.**
(density act egually on C_{lift}/C_{drag} ratio **LIFT = +(½ρ V^2) S C_L / DRAG = +(½ρ V^2) S C_D**

The frontal area of a **body**, placed in a certain airstream is **increased by a factor 3**. If the shape does not alter, the **form drag** will **increase by a factor of 3** (if density, speed or area change, the Lift or Drag change, but L/D ratio is always the same **+LIFT = (½ρ V^2) S C_L/+DRAG = (½ρ V^2) S C_D**)

Comparing the **lift coefficient and drag** coefficient for conventional aeroplanes at **normal angle** of attack **C_L is much greater than C_D.** **(airplane is made to have more lift than drag at 4° AoA)**

If the **wing area is increased**, **lift** will **increase because it is directly proportional to wing area.**

A body is placed in a certain airstream. If the **density** of the airstream **decreases** to **half (/2)** its original value, the **parasite drag** will **decrease** by a factor of **2**
(if density is halved q_2=(½ρ)/2 means that Drag is /2 → (½ρ/2 $\cancel{V^2 S C_D}$)

Regarding the lift formula, if **density doubles**, lift will **also double. 2 x (½ρ) = LIFT2**
Regarding the lift formula, if **airspeed doubles**, lift will be **4 times greater**. **2 x (V^2) = LIFT4**

Both **lift** and **drag** of an aerofoil are **proportional to the square of the velocity of the relative airflow.**

Variables required to calculate **drag** using the drag formula are **dynamic pressure, drag coefficient and wing area.** **DRAG = ($\frac{1}{2}\rho V^2$) S C_D**
Variables are required to **calculate lift** from the lift formula **Dynamic pressure, lift coefficient and wing area.** **LIFT = ($\frac{1}{2}\rho V^2$) S C_L**

Minimum drag of an aeroplane in straight and level flight occurs at the **maximum C_L / C_D ratio.**
At the highest value of the lift/drag ratio the total drag is lowest.
(Maximum Lift respect Drag means drag is less)

The lift **coefficient (C_L)** is a function of
Camber of the aerofoil section.
Angle of attack of the aerofoil section.

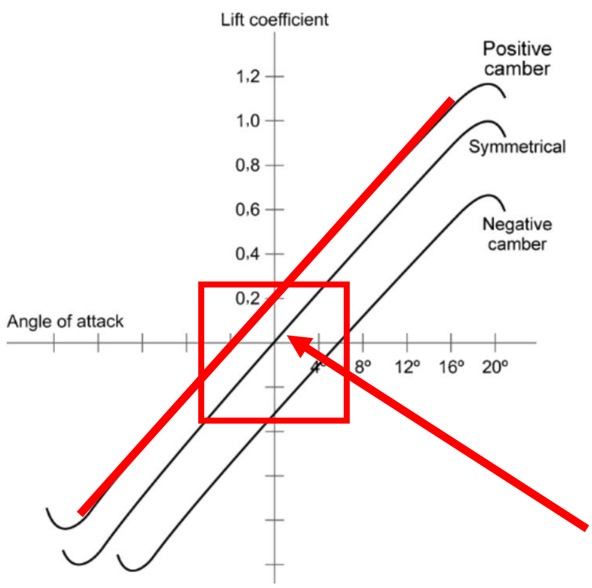
For aircraft of the same weight, flying at the **same IAS** the angle of **attack will** be **the same** at altitude as at sea level. (**IAS** = dynamic pressure, if pressure is **constant**, **angle is constant**)

If an aeroplane **maintains** straight and level flight at the same **angle of attack** at two **different altitudes**, the **TAS is higher at the higher altitude.**

If **TAS is kept constant**, to maintain straight and level flight with **reduced air density** the **angle of attack** of an aircraft's wings must be **increased.**

Increasing air pressure (angle of attack, OAT and TAS are constant) **the drag increases.** **+($\frac{1}{2}\rho V^2$)**
Increasing air density (angle of attack and TAS are constant) **the drag increases.** **+($\frac{1}{2}\rho$)**

The **lift** coefficient of an aerofoil section **increases with an increase in angle of attack up to the stall.**



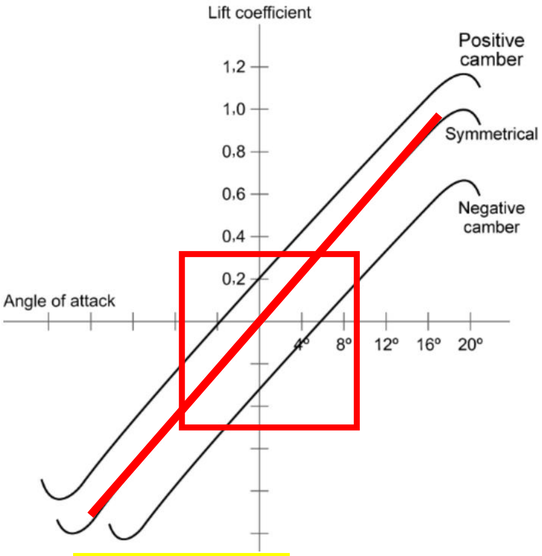
In a **positively cambered** aerofoil section **the angle of attack has a negative value when the lift coefficient equals zero.**
There is a nose down pitching moment about a positively cambered aerofoil when the lift coefficient equals zero.

Considering a **positively cambered** aerofoil section, the pitching moment when the **lift** coefficient **C_L=<0** is **negative.**

A **positively cambered** aerofoil will generate **zero lift** at a **negative angle of attack**.

A **positively cambered** aerofoil **it will generate lift at small negative angles of attack.**

The lift coefficient C_L versus angle of attack curve of a **positive cambered** aerofoil section intersects the vertical axis of the graph **Above the origin.**
The lift coefficient C_L versus angle of attack curve of a **positively cambered** aerofoil section intersects the horizontal axis of the grap **to the left of the origin.**



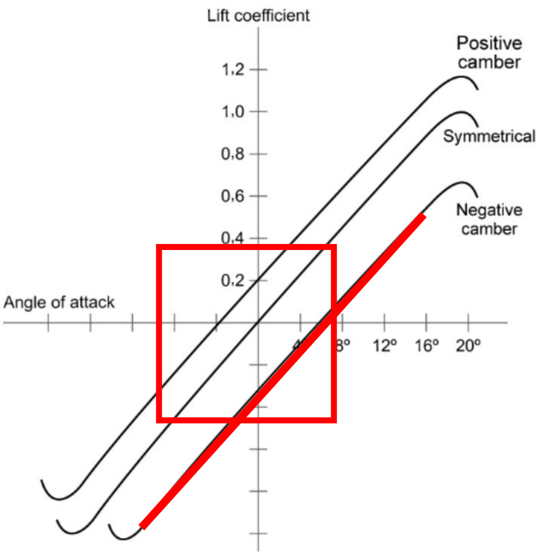
The **lift coefficient** of a **symmetrical** aerofoil section at zero angle of attack is **zero**.

In a **symmetric aerofoil** section **the angle of attack is zero when the lift coefficient equals zero.**
The pitching moment is zero when the lift coefficient equals zero.

When the **lift** coefficient C_L of a symmetrical aerofoil section **is zero**, the pitching **moment** is **zero**.

The **lift coefficient** C_L versus angle of attack curve of a **symmetrical** aerofoil section intersects the horizontal axis of the graph **at the origin**

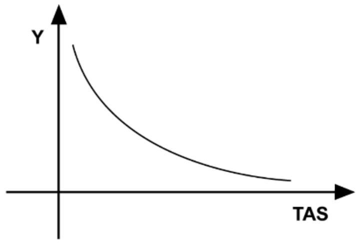
The **forces** acting on a **symmetrical** aerofoil element when the upstream airflow is parallel to the chord line **is only drag. (zero lift, but form and skin drag are present)**
For a symmetrical aerofoil section, if the angle of attack is zero, C_L is zero.
Symmetrical aerofoil at zero AoA produce Drag, but no lift



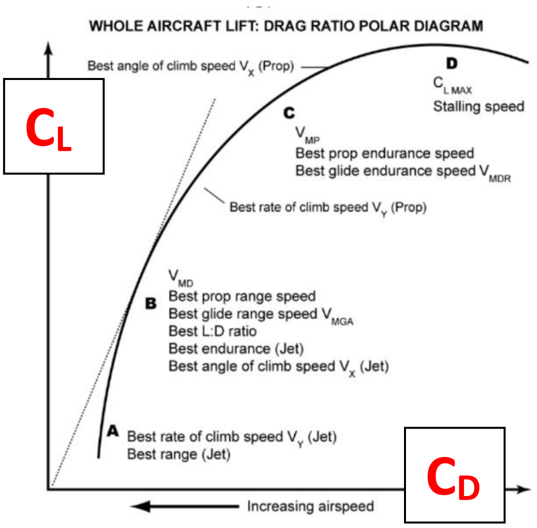
The **lift coefficient** C_L **versus angle of attack** curve of a **negatively cambered** aerofoil section intersects the vertical axis of the graph **below the origin** and the **horizontal axis** to **the right of the origin** (Generate less lift and negative lift at zero angle)

When the **lift coefficient** C_L of a **negatively cambered** aerofoil section **is zero**, the **pitching moment** is positive (pitch up).

For a given angle of attack, a **swept wing** will **have a lower coefficient of lift than an equivalent straight one.**
A **swept wing** compared to the same wing without sweep will provide **less lift** at a given angle of attack.
To maintain the same aircraft speed, a **swept wing** compared to the same wing without sweep will require a **higher angle of attack.**

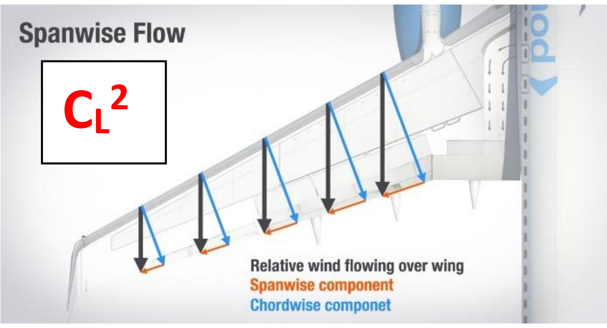


The diagram shows the parameter Y against TAS. If **horizontal flight** is considered axis Y represents the **lift coefficient.** **(+tas needs -AoA)**



The parameters that can be read from the "**parabolic polar**" curve of an aeroplane are **the minimum glide angle and the parasite drag coefficient.**
The **aerofoil polar** is a graph of the relation between the **lift coefficient** and the **drag coefficient.**
From the **polar curve** of the entire aircraft one can read the **maximum C_L/C_D ratio** and **maximum lift coefficient.**
The **polar curve** of an aerofoil is a graphic relation between **lift coefficient C_L and drag coefficient C_D.**
In the **polar curve** there are coefficient only, **C_L vs C_D**

Three-dimensional Airflow About an Aeroplane

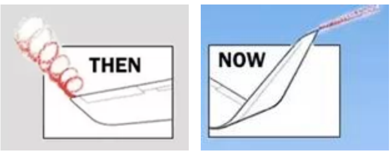


When wing **lift is zero**, its **induced drag is zero**
(induced = vortices – C_L^2 / Aspect ratio)
Excluding constants, the coefficient of **induced drag (C_{Di})** is the ratio of **C_L^2 and AR (aspect ratio).**
The **induced drag** coefficient, **C_{Di} is proportional with C_L^2.**
The **induced drag** - if speed remains constant – is **proportional to the square of the lift coefficient.**
An aeroplane fly into a horizontal co-ordinated turn with a **load factor of 2**, the speed remains constant and the **induced drag increases by a factor of 4.**
(The induced drag coefficient C_{Di} is equal to C_L^2, if Lift is 2, induced drag is 4)

The **difference** between the aerodynamic characteristics of **two and three-dimensional flow** is that, in a three-dimensional flow about a wing **a spanwise component exists in addition to the chordwise speed component.**
Induced drag is created by the **spanwise flow** pattern resulting in **the tip vortices**.

The strength of wing tip vortices increases as the aspect ratio decreases. (more shorter is the wing more vortices create. **High aspect ratio means thin and longer wings**)
Increasing the **aspect ratio** of a wing **decreases induced drag.**

On an **untampered** wing of constant section **induced downwash is maximum at the wing tip.**



By fitting winglets to the wing tip, the strength of the wing tip vortices is reduced which in turn reduces induced drag
Winglets decrease the **induced drag.**

Wing tip of the aeroplane has the largest effect on **induced drag**

Induced drag is the result of **downwash generated by tip vortices.**
Induced drag is caused by **wing tip vortices and downwash.**

The effect of **winglets** on the drag of the wing is **Increase parasite drag, decrease induced drag.**

The **wing tip** vortices behind a large aircraft begin on take-off **on rotation** and end on landing as the **nosewheel touches down**

Vortex wake behind large aircraft **gradually descends to a lower level.**

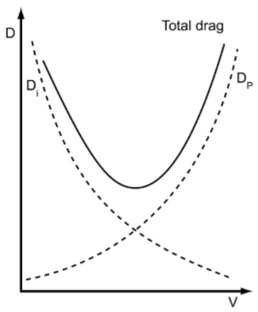
Assuming zero wing twist, the wing planform that gives the **highest local lift** coefficient at the **wing** root **is rectangular.**

Assuming no flow separation, the strength of wing tip vortices increases as the angle of attack increases.
Assuming no flow separation, the strength of wing tip vortices decreases as the angle of attack decreases.
The strength of wing tip vortices increases as the aspect ratio decreases.

The effect of increasing **aspect ratio** is to **increase the maximum lift / drag ratio.**

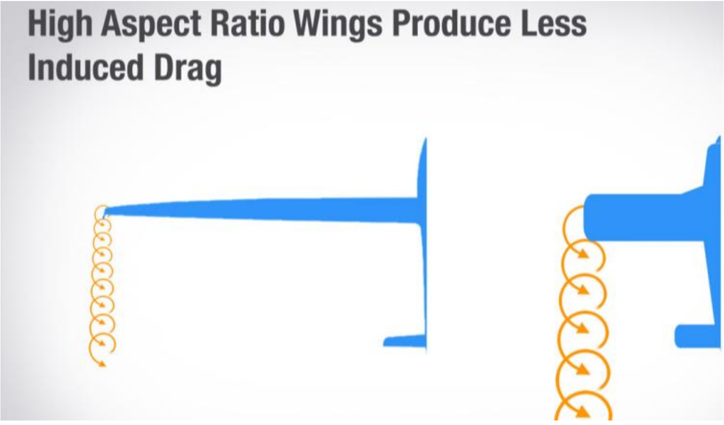
Decreasing the aspect ratio of a wing **increases induced drag.**

Increase in aspect ratio Induced drag decreases, because the effect of tip vortices decreases.
A **decrease in the aspect ratio increases the induced drag.**



Speed increases, induced drag decreases.
Speed decreases, induced drag increases.
Mass increases, induced drag increases.
Mass decreases, induced drag decreases.
Angle of attack increase, induced drag increase
Angle of attack decrease, induced drag decrease

The effect on **induced drag** of mass and speed changes are **decreases with increasing speed and decreasing mass.**
As **fuel is consumed** during a level flight cruise at high level **induced mass decreases**



For a **rectangular wing** at constant speed and angle of attack, **induced drag will be greatest at the wing tip.**

A **rectangular spanwise** lift distribution **generates more induced drag than an elliptical lift distribution.**

For a **rectangular wing the flow on the upper surface of the wing has a component in wing root direction.**

An **elliptical spanwise** lift distribution generates **less induced drag** than a rectangular lift distribution **(elliptical = medium aspect ratio → spitfire)**

Elliptical wing planforms produces the **lowest induced drag**

Induced drag decreases with increasing aspect ratio.

Induced drag may be reduced by an **increase in aspect ratio.**

High aspect ratio decreasing induced drag and **critical angle of attack.**

Increasing wing aspect ratio, induced drag It is reduced because the effect of **wing-tip vortices is reduced.**

The **induced angle of attack** is the result of **downwash due to tip vortices.**

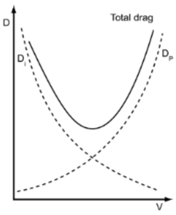
The **induced angle of attack** is the angle by which the **relative airflow is deflected** due to **downwash.**

The **induced drag increases as the lift coefficient increases.**

The value of the **induced drag** of an aeroplane in straight and level flight at a constant weight **varies linearly with $1/V^2$ (decrease with increasing speed)**

Assuming no compressibility effects, **induced drag** at **constant IAS** is affected by **aeroplane mass.**

The **nose up or nose down** orientation of an aircraft relative to the horizon is known as **the attitude of the aircraft.**



The effects on drag of **fitting external tip tanks** to the wings of an aeroplane **parasite drag increases.**

The effects on drag of **removing external tip tanks** from the wings of an aeroplane are **Induced drag increases.**

With **flaps deployed**, at a constant IAS in straight and level flight, the magnitude of **tip vortices decreases.** **(Flap generate more parasite drag than lift)**

Wing tip vortices have the greatest intensity **during take-off.**

The **span-wise flow (vortex)** on an unswept wing is from the **lower to the upper wing surface via the wing tip.**

For a **tapered wing without twist**, the effective **angle of attack** will be **greatest at the root.**

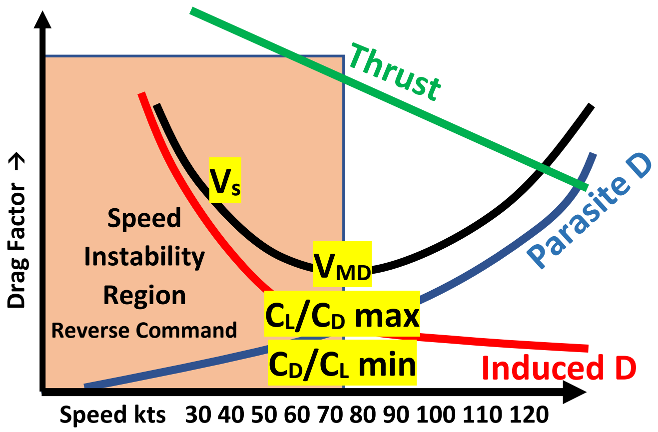
Wing twist (wash out) (geometric and aerodynamic) is used to
improve stall characteristics.
reduce induced drag.

Total Drag

$$\text{DRAG} = \tfrac{1}{2}\rho\, v^2\, S\, C_D$$

Total Drag = **Induced drag (vortices)** +
Parassite Drag (Pressure drag + Friction drag + Interference drag.)

The **aerodynamic drag** of a body, placed in a certain airstream depends amongst others on **the airstream velocity.**
The value of the **parasite drag** in straight and level flight at constant weight **varies linearly** with the **square of velocity** Parasite **DRAG** = $\cancel{\tfrac{1}{2}\rho}\, v^2\, \cancel{S}\, C_D$



The **total drag** of a three dimensional wing consists of **induced drag and parasite drag.**

The **total drag** in straight and level flight is **lowest** when the **parasite drag is equal to the induced drag.**
The **minimum total drag** of an aircraft in flight occurs **at the speed where parasite drag and induced drag are equal.**

As speed is increased from the stall speed (V_S) to maximum IAS (V_{NE} or V_{MO}), in straight and level flight at constant mass, **the total drag** change **Initially decreases, then increases.**

With **increasing speed** in straight and level flight **induced drag decreases and parasite drag increases.**

At **high speed** an aircraft will have **more profile drag** than induced drag.

Speed stability of an aircraft **is stable above V_{MD} because total drag increases as speed increases.**

The **speed instability region(reverse command)** is the **same** as the **region of negative gradient of the thrust or power required curve**
The **region of speed instability** is **the same as the region of reversed command.**

Important aspects of the "**backside of the power curve** is that **the speed is unstable.**

As **speed** is reduced **below V_{MD} drag increases and speed stability decreases.**
If the **airspeed reduces** in level flight **below the speed for maximum L/D**, the **total drag** of an aeroplane will **increase because of increased induced drag**.

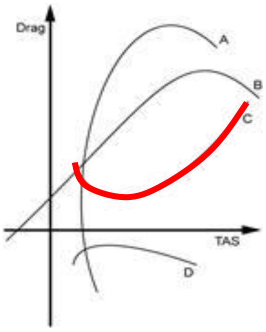
The **resistance, or skin friction**, due to the viscosity of the air as it passes along the surface of the wing **is part of** the **parasite drag.**

The **interference drag** is the result of **aerodynamic interaction between parts of the aeroplane (e.g. landing gear/fuselage).**

An aeroplane in the **speed unstable region**, experiences a speed reduction. If the **altitude is maintained and thrust remains constant**, the aeroplane speed **will further decrease (unstable = further decrease or increase according to the disturbance)**

An aircraft is said to have **speed stability when the speed is disturbed from its trimmed value, it tends to return to the original speed.**

Increasing dynamic pressure (speed) **at speeds above the minimum drag speed, total drag increases**.



The **total drag line** of an aeroplane is **line C**

At a **load factor of 1** and the aeroplane's **minimum drag speed**, **the ratio** between induced drag D_i and parasite drag D_p is **D_i / D_p = 1, because at minimum total drag speed they are equals**

If the **weight of an aircraft is increased**, for a **constant speed induced drag will increase; profile drag will remain the same.**

When aeroplane **speed is doubled**, drag increase by a factor of **4** **DRAG = ½ρ 2^2 S C_D**

An aeroplane accelerates from 120 kts to 160 kts in level flight. The Parasite drag changes by a factor of **1,78** and the Induced drag coefficient changes by a factor of **0,32**

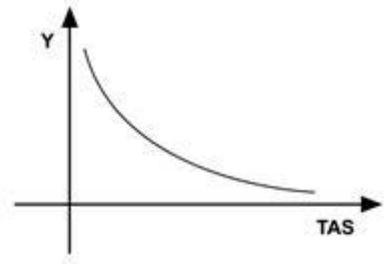
(120 ÷ 160 = 100% ÷ x = **133,34% → 33,34%**
Speed increase by a factor of $1{,}3334^2$
parasite drag increase proportionally → **DRAG = $\frac{1}{2}\rho$ $1{,}334^2$ S C_D = 1,78)**

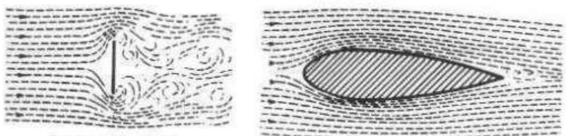
A body is placed in a certain airstream. If the airstream **velocity increases by a factor of 4**, the **parasite drag will increase by a factor of 16** → **DRAG = $\frac{1}{2}\rho$ 4^2 S C_D**

An aeroplane accelerates from 80 kts to 160 kts at a **load factor equal to 1**. **The induced drag coefficient will change by 1/16 and the induced drag will change by 1/4.** (*we must double the speed and use a value of 1 for the rest → DRAG = $2V^2$= 4*
Drag in this case is 4 times bigger than coefficient of drag or 1/4 vs 1/16)

If the speed during a straight and level flight is increased by a factor of 2, the parasite drag change by a factor of **4** **(DRAG= $2V^2$ = 4)**

If horizontal flight is considered, axis Y represents **the induced drag.**
(induced drag decrease with increasing speed)





Resistance, 100%

Resistance, 5%

The purpose of **streamlining is to reduce form drag.**
(streamlining = thinner, reduce wings thickness)

Profile drag is **directly proportional to the square of the speed.** **DRAG = $\cancel{\frac{1}{2}\rho}$ V^2 $\cancel{S\,C_D}$**

Ground Effect

Ground effect increase lift (C$_L$ & Effective AoA) reduce vortices (drag, induced, downwash)

an aeroplane **leaving ground effect** (less lift)
induced angle of attack increase.
induced drag coefficient C_{Di} increase.
downwash angle of attack increase
lift coefficient C_L decrease.
Effective angle of attack decrease **(angle relative to airflow)**

	↓	↑
D	-	+
L	+	-
D	-	+
E	+	-
I	-	+

About an aeroplane **entering ground effect** (more lift)
induced angle of attack decrease.

induced drag coefficient C_{Di} **decrease**.
downwash angle of attack **decrease**.
lift coefficient C_L increase.
Effective angle of attack increase **(angle relative to airflow)**

Entering = Effective Cl ↑

In **ground effect** the **lift is increased and the drag is decreased.**
In **ground effect** the **effective angle of attack is increased**
In **ground effect** the **induced drag reduce.**
In **leaving ground effect** the **lift is decreased and the drag is increased.**
In **leaving ground effect** the **effective angle of attack is decreased**
In **leaving ground effect** the **induced drag increase.**

Floating due to **ground effect** during an approach to land will occur when **the height is less than half of the length of the wingspan above the surface.**

A pilot must be aware of as a result of **ground effect** because **Induced drag decreases, and any excess speed in the flare may cause floating.**

With **Ground effect** the **landing distance increases.**

Ground effect is most likely to result in **Becoming airborne before reaching recommended take-off speed.**

low wing & low tailplane constellation will have the greatest effect on the **ground effect**

Relationship between the C_L and Speed in Level Flight

If No compressibility effect an aeroplane maintains **straight and level flight at the same angle of attack**:
TAS is lower at lower altitude.
TAS is higher at higher altitude.
IAS at different altitudes is the same. (No compressibility = disregard density. Ias is calculated from density)

Assuming **standard atmospheric conditions**, in order to generate the **same amount of lift** as **altitude is increased**, an aeroplane must be flown at **a higher TAS for any given angle of attack.**

When you **decrease the angle of attack velocity increase and lift coefficient decrease**

If the **airspeed** is being **decreased** airplane must **Increase the angle of attack to compensate for the decreasing lift.**

If the **airspeed is doubled**, whilst **maintaining the same control surface deflection** the **aerodynamic force** on this control surface will **become four times greater. LIFT = $V2^2$ = 4**

An aeroplane maintains straight and level flight while the **speed is doubled**.
The change in **lift coefficient will** be **× 0.25** (**LIFT = ½ρ 2^2 S C_L → 2^2 x C_L** → 4 so 1/4 = 0,25)
The change in **lift factor will be 4**

In straight and level flight at a **speed of 1.1 V_S**, the **lift coefficient**, expressed as a **percentage of its maximum (C_{LMAX})**, would be **83%** (**LIFT = ½ρ $1{,}1^2$ S C_L → $1{,}1^2$ x C_L** → 1,21. Increased by a factor of 1,21 but question ask percentage respect its maximum C_{LMAX} so 1/1,21 = 0,83)

In straight and level flight at a **speed of 1.8 V_S**, the **lift coefficient**, expressed as a **percentage of its maximum (C_{LMAX})**, would be **30%** (**LIFT = ½ρ $1{,}8^2$ S C_L → $1{,}8^2$ x C_L** → 3,24. Increased by a factor of 3,24 but question ask percentage respect its maximum C_{LMAX} so 1/3,24 = 0,31)

In straight and level flight at a **speed of 1.3 V_S**, the **lift coefficient**, expressed as a **percentage of its maximum (C_{LMAX})**, would be **59%** (**LIFT = ½ρ $1{,}3^2$ S C_L → $1{,}3^2$ x C_L** → 1,69. Increased by a factor of 1,69 but question ask percentage respect its maximum C_{LMAX} so 1/1,69 = 0,59)

maintaining straight and level flight with a lift coefficient C_L = 1 after the **speed is increased by 30%** the new approximate value of C_L would be **60%** (**LIFT = ½ρ $1{,}3^2$ S C_L → $1{,}3^2$ x C_L** → 1,69. Increased by a factor of 1,69 → so 1/1,69 = 0,59)

maintaining straight and level flight with a lift coefficient C_L = 1 after the **speed is increased by 41%** the new approximate value of C_L would be **50%** (**LIFT = ½ρ $1{,}41^2$ S C_L → $1{,}41^2$ x C_L** → 1,99. Increased by a factor of 1,99 → so 1/1,99 = 0,50)

maintaining straight and level flight with a lift coefficient C_L = 1 after the **speed is doubled** the new approximate value of C_L would be **0,25** (**LIFT = ½ρ 2^2 S C_L → 2^2 x C_L** → 4. Increased by a factor of 4 → so 1/4 = 0,25)

An aeroplane is **in level flight at 146 kts** with a **Lift coefficient (C_L) of 0.91**. If the 1g **stall speed is 109 kts**, the value of C_{LMAX} is **1,63** (½ρ x **146^2** x S x **C_L0.91** = ½ρ x **109^2** x S x **C_{Lmax}** → 19397,56 = 11881 x C_{LMAX} → 19397,56 / 11881 = C_{LMAX} **1,63**)

$146^2 \times 0.91 / 109^2 = 1.63$

The Stall (Stall=Critical angle of attack)

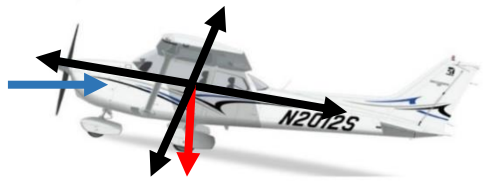
Stall = 1 / cos(°)

V_s = V x √LOAD FACTOR

V_s = 1 ÷ √cosΦ

V_s higher = V_S **lower × √(+mass ÷ mass)**

New old new old



The **load factor** is **lift / weight**

$V_S = 109$ LF 1.8 New V_S? $109 \cdot \sqrt{1.8} = 146,2$

If the **stall speed** of an aeroplane is 60 kts, the **load factor** is 2, the stall speed will be **85 kts.**
(V_S = SPEED x √LOAD FACTOR → 60 x √2 = 85)

An aeroplane has a **stalling speed** of 100 kts in a steady level flight. When the aeroplane is flying a level turn with a **load factor** of 1.5 the stalling speed is **122 kts**
(V_S = SPEED x √LOAD FACTOR → 100 x √1,5 = 122)

The stall speed in **a 60° banked turn** increases by the following factor **1,41**
(V_S = 1 ÷ √cos Φ → 1,41)

An aircraft maintains a **50° bank angle** during a constant altitude turn. Compared to straight and level flight the **stall speed will be greater** by a factor of **1,25** **(V_S = 1 ÷ √cos Φ → 1,25)**

Given an initial condition in straight and level flight with a speed of 1.4 V_S. The **maximum bank angle** attainable without stalling is **60°** (Try all answer, 44°, 32°, 30°, 60°. **V_S = 1 ÷ √cos 60° → 1,4)**

Stalling speed in a 15° bank level turn is 60 kts. The stalling speed in a 45° bank level turn ? is **70 kts**
(V_S = 1 ÷ √cos Φ → 60 kts × √(cos 15° ÷ cos 45°)= 70kts

An aeroplane has a stall speed of 78 kts at its mass of 6 850 kg. The stall speed when the mass is reduced to 5 000 kg is **67kts** **(V_S lower** = **V_S higher** **× √(lowerKG ÷ higherKG)).**
New Old New Old

An aeroplane has a stall speed of 100 kts at a mass of 1 000 kg. If the mass is **increased** to 2 000 kg stall speed is **141 kts** **(V_S higher** = **V_S lower × √(higherKG ÷ lowerKG)).**

Load factor directly affects stall speed and is the **ratio** between the total air **load imposed on the wing** and the gross **weight of an aircraft** in flight
The **load factor is more than 1** (one) **when lift is more than weight.**
The **load factor is less than 1** (one):**during a steady climb.**
The **load factor is less than 1** (one):**lift is less than weight.**
The **load factor is less than 1** (one):**during a steady wings level descent.**
The **load factor is egual to 1** (one) **in steady level horizontal flight**

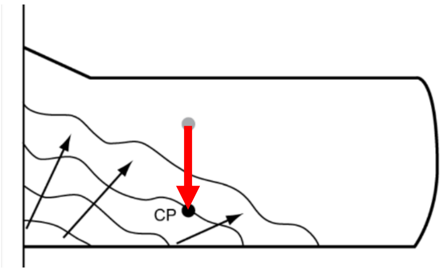
As the centre of gravity is changed, **recovery from a stall** becomes progressively **more difficult** as the **centre of gravity moves aft.**

Comparing the IAS and TAS stall speed at 5000 ft at the **higher altitude** the **IAS stalling speed** will be **the same as at sea level but the TAS will be higher.**
At very **higher** altitudes, the **stall speed (IAS)**: **increase (due to compressibility effects.)**
Stall speed IAS varies only with **weight at low altitude**.
As **altitude is increased**, the **stall speed (IAS) initially remains constant and at higher altitudes increases.**
Stalling speed (IAS) It remains constant at lower altitudes but increases at higher altitudes due to compressibility effects.
When a pilot performs a steady **horizontal turn**, the **stall speed increases** with the **square root of load factor.**

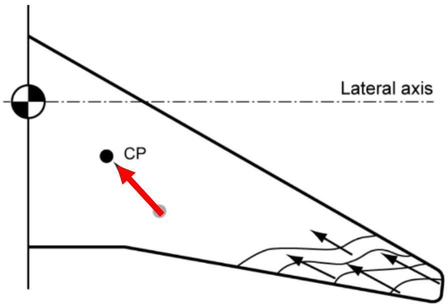
The **stall speed (IAS)** may be **increased** by **high altitude; icing conditions; forward movement of the CG.**
The **stalling speed (IAS) may increase during turbulence and will always increase when banking in a turn.**
The **stalling speed (IAS) increase during turn, increased mass and forward CG location.**
With an increase in load factor, a forward CG shift, decrease in thrust the stall speed **increase**
The **stall speed increase** when **pulling out of a dive**. (return in level flight increase V_s to normal)
When **pulling out of a dive** the **angle of attack increases**.

The **stall speed decreases when, during a manoeuvre, the aeroplane nose is suddenly pushed firmly downwards (e.g. as in a push over).**
The **stall speed (IAS) decreases when Decreasing weight.**

The **TAS** of an aircraft at the stalling angle of attack at a given weight **increases as altitude increases.**



About the **stall of a straight wing** aeroplane **just before the stall the aeroplane will have a nose-down tendency.**
A **rectangular wing**, compared to other wing planforms, has a tendency to **stall first at the wing root, with the stall progression towards the wing tip.**



On a **swept wing** aeroplane at low airspeed, the **pitch up** phenomenon **is caused by wingtip stall.**
Low speed pitch up is caused by the spanwise flow on a **swept back wing**.
The **pitch up tendency** of an aeroplane with **swept back wings** during a stall is caused by the **forward movement of the centre of pressure.**
On a **swept wing** aeroplane **tip stall will occur first, which produces a nose-up pitching moment.**
The **pitch up** effect of an aeroplane with **sweepback** wings in a stall is due to the **wing tip stalling first.**

Low speed pitch up can be caused by the **outward drift of the boundary layer on a swept-back wing.**
Wing sweep has a **negative influence on stall characteristics as it increases stalling speed and could cause tip stall.** One disadvantage of wing **sweepback** is **the tendency of the wingtip section to stall prior to the wing root section.**
When sweep back increases the centre of pressure has an increased tendency to move forward as the angle of attack approaches and exceeds the critical angle of attack.
Decreasing forward sweep decreases stall speed.
Increasing sweepback increases stall speed.
Decreasing sweepback decreases stall speed.

Cessna

The centre of pressure on a straight wing/unswept moves aft as the angle of attack approaches and exceeds the critical angle of attack. Airbus
The centre of pressure on a strongly swept back wing moves forward as the angle of attack approaches and exceeds the critical angle of attack.

The **critical angle of attack** is **dependent** on the **design of the wing.**

With **increasing angle of attack** the **CP** will reach its most forward point **just below the stalling angle.**

During a normal **recovery of a spin the ailerons are held in the neutral position.**
During spin recovery the ailerons should be kept in the neutral position. (use rudder!!!)

If the **aspect ratio of a wing increases**, the critical angle of attack will **decrease**.
Fighter jets, low aspect ratio and a high crit AoA, **gliders high AR but low crit AoA**

During a steady horizontal **turn**, the **stall speed increases with the square root of the load factor.**

In a steady, level, co-ordinated **turn** the load factor **"n" greater than 1** and the **stall speed V_S higher** than in straight and level flight.

Increase of **wing loading** will **increase the stall speeds.**

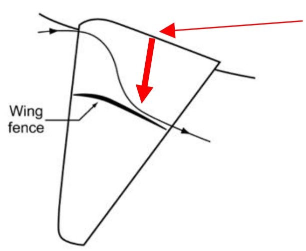
The **vane of a stall warning** system with a flapper switch is **activated** by the **change of the stagnation point.**
As the stagnation point moves downwards the flapper switch moves upwards and the **stall warning** is triggered in an aircraft.
The **sensor** of a **stall warning** system can be **activated** by a change in the location of the **stagnation point.** A **stall warning vane** on an aircraft wing is fitted **just below the leading edge.**

Compared with **stalling airspeed** in a given configuration (V_S), the **airspeed at which stick shaker** will be triggered is **greater than V_S. (stick shaker advise you before reaching the V_S)**

The input to a stick shaker comes from angle of attack, and sometimes the rate of change in angle of attack.
If aeroplane slowly decelerates in level flight from its cruise speed in still air at high altitude stick shaker activation or low speed buffeting appear.

A stick pusher activates at a higher angle of attack than a stick shaker. (first activate the shaker, than the pusher)
A stick pusher activate and push the stick forward at or beyond a certain value of angle of attack.
Dangerous stall characteristics, in large transport aeroplanes that require stick pushers to be installed, are excessive wing drop and deep stall.
A stick pusher pushes the elevator control forward when a specified value of angle of attack is exceeded.

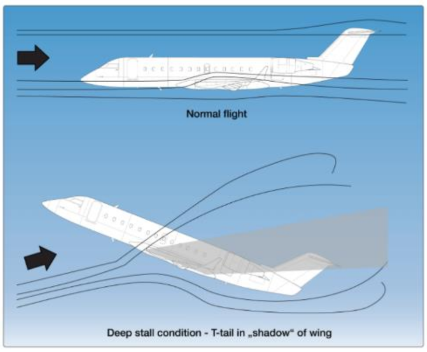
A jet aeroplane cruises buffet free at constant high altitude. If the aeroplane decelerates during an inadvertant increase in load factor there is a possibility of accelerated stall.
(Accellerated stall = High angle of attack at Hight speed or High load factor)



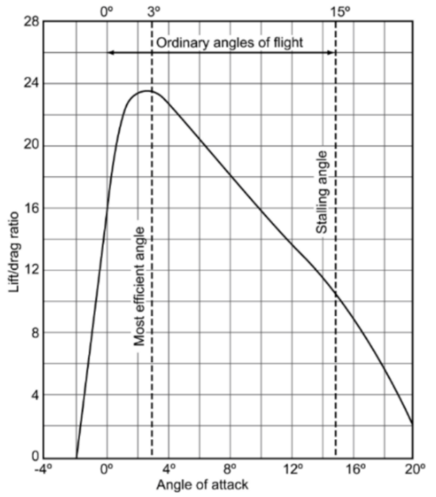
It is possible to reduce the spanwise airflow over swept wings, due to adverse pressure gradients, by wing fences.
The main purpose of a boundary-layer fence on a swept wing is to improve the low speed handling characteristics.
(fence reduce wing tip stall by maintaining the airflow close to the wing root)

A fixed stall strip/spoiler on the leading edge of the wing at the root will induce a root stall
The purpose of a fixed spoiler/strip on the leading edge of a wing at the root is to ensure that the root of the wing stalls before the tip.

Interactive Broker



The highest probability of a super stall is with Swept wings
Super stall is first with Swept wings than with a T-Tail
Super stall is a condition which is a stable stall with almost a constant pitch attitude.
Deep stall is cause by swept back wings and a T-tail.
The type of stall that has the largest associated angle of attack is Deep stall. (30°-40°)
Negative tail stall is a sudden reduction in the downward aerodynamic force on the tailplane.
When negative tail stall occurs, the aeroplane will show a pitch up moment but an uncontrollable pitchdown moment. (angle of attack is higher and wind push the airplane back)
The combination of a wing with sweepback and a T-tail make an aeroplane prone to deep stall.
The combination of a sweepback and a low horizontal tail make an aeroplane prone to deep stall.
A stick pusher system is fitted to an aeroplane to resolve deep stall problems.
A stick pusher system can be fitted to an aeroplane that exhibits abnormal stall characteristics.



The **lift/drag ratio** reaches its **maximum value at the peak of the lift/drag curve.**

The **stalling angle** of an aerofoil is approximately **16°**
The **critical angle** of attack **remains unchanged** regardless of gross weight.
The **stalling angle** of a wing is **unaffected by a turn.**

When an aerofoil is **stalled** the **lift/drag ratio is less** than the most efficient ratio.

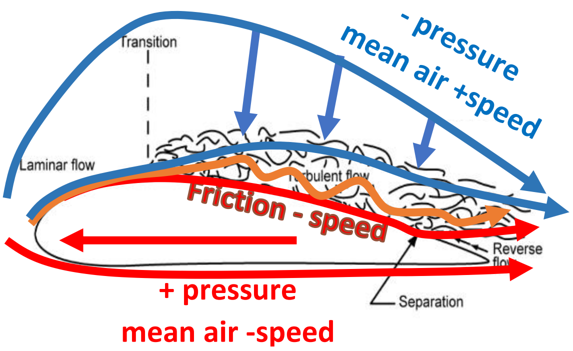
The **stall speed increases with an increased weight.**
A **decrease in weight** due to fuel consumption in flight will **reduce the stalling speed, but the stall angle remains the same.**

At the same weight, with the **CG at its forward limit V_S is higher,** the **stalling angle is unchanged.**

Stall speeds are determined with the CG at the forward limit.
Minimum control speeds are **determined** with the **CG at the aft limit.**

Vsr is the **Reference stall speed**
Vso is the **stall speed in landing configuration**
Vs1g is the **minimum speed at which the aeroplane can develop a lift force (normal to the flight path) equal to its weight. Vs1g lift equals weight**
Vs1 is the **the stall speed or the minimum steady flight speed obtained in a specific configuration.**

VS1G when lift=weight – VS1 in a specific configuration



The separation (reversal) point is where **airflow leaves the surface** of an aerofoil
Airflow separation is characterised by **airflow reversal on the surface of the body.**

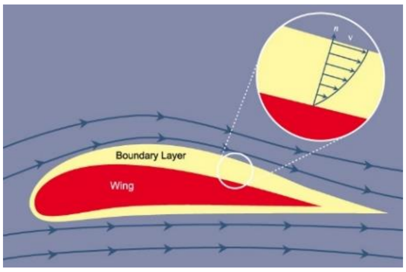
The transition point is where a laminar boundary layer becomes **turbulent**

The **airflow over the wing** from the **wing surface and up** is:

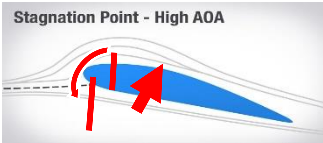
1) **accelerated to the transition point (becomes turbulent)**
2) **accelerated from transition point to separation point**
3) **after separation point/reversal leaves the surface**

As angle of attack is increased, **flow separation** normally starts on the **upper surface near the trailing edge.**
In the stall of the wing the lift decrease and drag increase

(vortex and separation of laminar flow increase drag)



The **boundary layer** (skin friction layer) of a wing is caused/created by **a layer on the wing in which the stream velocity is lower than the free stream velocity.**

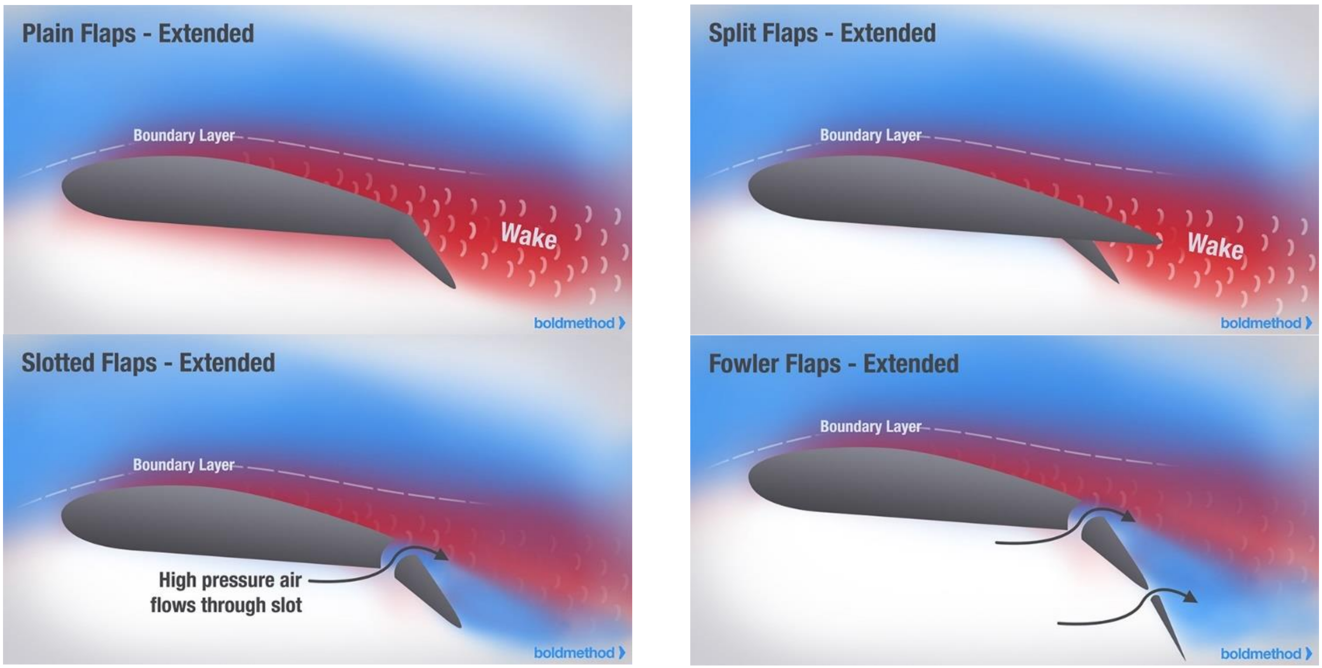


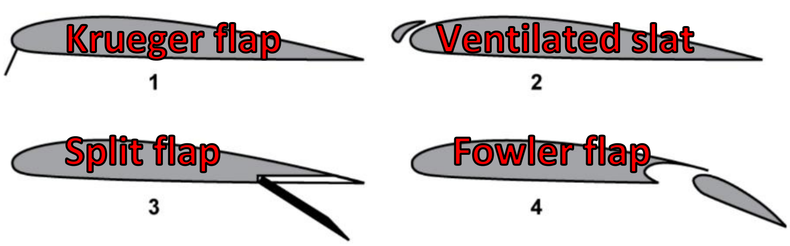
In **critical angle of attack**, the **lift coefficient decreases** and the stagnation point moves **rearward**

During a **spin** to the **left, both wings are stalled**

C_{LMAX} Augmentation

The C_{LMAX} is the maximum C_L that can be generated from that aerofoil section and it is stabilized during the designment. Since reference area S of an aerofoil is fixed the only way to increase C_{LMAX} Is to change the area using FLAP/SLATS.

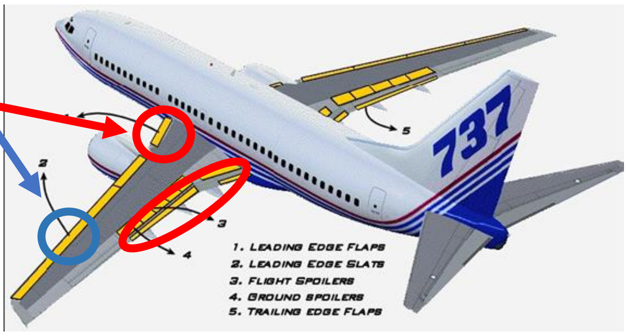




A **slotted** flap will **increase the C_{LMAX}** by **increasing the camber of the aerofoil and re-energising the airflow.**

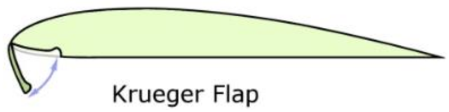
A **plain flap** will **increase C_{LMAX}** by **increasing the camber of the aerofoil.**

On a **highly swept back** wing with **leading** edge slats fitted outboard and **leading** edge flap fitted inboard



Flaps (leading/trailing) **and** slats are fitted to modern aircraft **to reduce take-off, approach and landing speeds to an acceptable level.**

The Krueger flaps are **leading edge flaps close to the** wing root.



A Krueger flap is part of the lower surface **of the leading edge, hinged at its forward edge.**

An aeroplane with swept back wings is equipped with slats and/or leading edge (LE) flaps. One possible efficient way to arrange the leading edge devices on the wings is **Wing roots: LE flaps; wing tips: slats.**

Lowering the inboard flaps causes the wing **CP to move inboard towards the wing root.**

When the trailing edge flaps are lowered to the fully extended position **the wing CP moves aft/rear and the L/D ratio decreases.**

When trailing edge flaps are extended whilst maintaining straight and level flight at constant IAS **the centre of pressure moves aft.**
Trailing edge flaps increase lift at a lower Angle of Attack.
Trailing edge flaps increase drag at higher Angle of Attack.
Trailing edge flaps **provide the same amount of lift at a slower speed.**
Trailing edge flaps extended **degrade the minimum glide angle.**

When the flaps are extended. **The negative angle of attack of the tailplane increases and the effectiveness of the tailplane increases.**
If the increased **downwash** at the **tail plane** due to lowering (extending) trailing edge **flaps** is considered in **isolation a nose up pitching moment will be generated.** (flap extended give nose down, but if we disregard this and we consider only the downwash on tail, increased stability of stabilizer, the tail is more effective causing a nose up)

From an initial condition of level flight the **flaps are retracted** at a **constant pitch attitude**. **The aeroplane will subsequently start to sink.**
When **trailing edge flaps** are extended, the **change in pitching moment**, will be **nose down**
The **flaps are extended**, the aeroplane will **subsequently start a climb**

During the **retraction of the flaps** at a constant angle of attack the aeroplane starts to **sink suddenly.**

The **highest** value of the **C_L/C_D ratio** is in **Clean configuration**.

When flaps are deployed the lift coefficient will increase (if not in level flight or same IAS)
When flaps are retracted the lift coefficient will decrease (if not in level flight or same IAS)

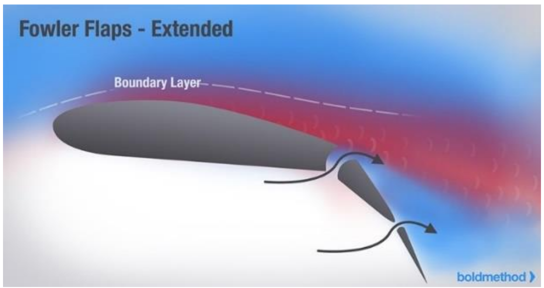
The **lift coefficient C_L** of a wing at a given angle of attack **is increased by the use of high lift devices.**

The purpose of correctly setting the **leading and trailing edge devices** on the wing of an aeroplane during take-off, approach and landing is to **reduce stall speed, increase C_{LMAX} with minimum increase in drag for take-off, but with a relatively high drag for approach and landing.**

Flap reduce critical angle of attack, Slat, Krueger Flap Increase critical angle of attack.

Compared with the clean configuration, **the angle of attack at C_{LMAX}** with **trailing edge flaps** extended is **smaller**
The **critical angle of attack** for the **flaps down** configuration, compared to flaps up is **smaller.**
When **flaps are extended C_{LMAX} increases and the critical angle of attack decreases.**
The main function of a trailing edge **flap** is to **increase** the maximum **lift coefficient** of the wing.
Trailing edge flap extension will **decrease the critical angle of attack and increase the value of C_{LMAX}.**

If **flaps** are extended slightly **asymmetrically** this would cause **a steady (continue) rate of roll which may be correctable with ailerons.**



When **Fowler flaps are extended** at a constant angle of attack **C_L and C_D increase.**
A **fowler** flap, fully extended position will produce **an increase in wing area and camber.**
Extending **Fowler flaps** to their fully **extended** position **wing area and camber increase.**
Fowler flaps extended a nose-down pitching moment. Deploying a **Fowler flap**, the flap will **move aft, then turn down. Fowler flap is the most effective one**

In order to **maintain straight and level flight** at a constant airspeed, whilst the **flaps are being retracted**, the **angle of attack is increased.**

If the **trailing edge flaps** are lowered (**fully extended**), in order to **maintain constant IAS and level flight the nose must be lowered and thrust increased.**

When **flaps are extended** whilst **maintaining straight and level flight** at **constant IAS**, the **lift coefficient** will **eventually remain the same.**

Flap extension maintaining straight and level flight will increase the **maximum lift coefficient (C_{LMAX}) and the drag.**
To maintain level flight and IAS, LIFT CONSTANT

Configurations in order of **increasing critical angle of attack (reducing stall speed)**
1) flaps only extended. (lower crit AoA)
2) clean wings.
3) slats only extended. (higher crit AoA).

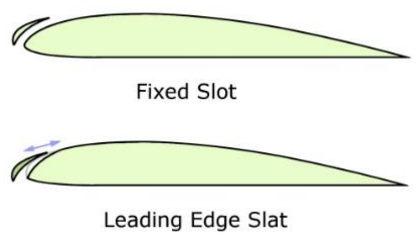
C_{LMAX} may be increased by the use of

C_{LMAX} (Max Lift) = Max C_L

- **boundary layer control.**
- **leading edge slats.**
- **trailing edge flaps.**
- **all of the above.**

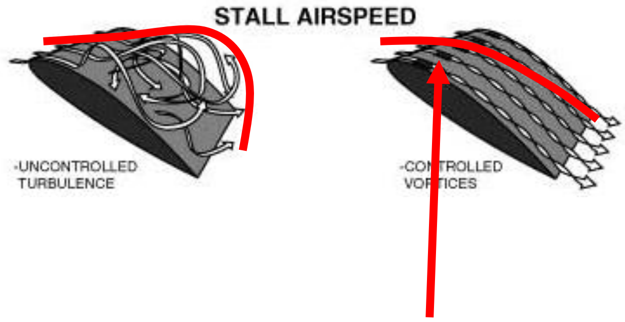
The selection that provides the **highest** positive contribution to C_{LMAX} is **slats from Retracted to Extended.**
The **difference** between the effects of **slat** and **flap** asymmetry is that **flap asymmetry** causes a **large (difficult) rolling moment** at any speed whereas **slat asymmetry** causes a **large difference in C_{LMAX}.**

Slat asymmetry causes a yawing moment, whereas flap asymmetry causes a large (difficult) rolling moment



Slat extension **delays the stall to a higher angle of attack.**
Deploying a slat will form a slot, deploying a Krueger flap does not.
Slat" is a leading edge device that causes some of the high energy air to flow over the upper surface of the wing.
For most jet transport aeroplanes, **slat** extension **has a greater effect on stall speed** than flap extension.
A deployed **slat** will **increase the boundary layer energy and increase the suction peak on the fixed part of the wing, so that the stall is postponed to higher angles of attack.** After take-off the **slats** (when installed) are always **retracted later** than the flaps **because slats extended give a large decrease in stall speed with relatively less drag.**
Slats re-energise the boundary layer thereby increasing the stalling angle of attack.
A "**Slat**" is an **auxiliary, cambered aerofoil positioned forward of the main aerofoil so as to form a slot.** The use of a **slot delays the stall to a higher angle of attack.**
The function of the **slot** between slat and the leading edge is to **cause a venturi effect which energizes the boundary layer.**
On a large transport aeroplane, **the auto-slat system extends** the slats **automatically** when a certain value of angle of attack is exceeded.
An **automatic leading edge slat** is operated by **aerodynamic forces acting on the leading edge.**
Extension of leading edge flap increases the **critical angle of attack.**
Extension of slat increases the **C_{LMAX}.**

An aeroplane has the following flap positions 0°, 15°, 30°, 45°; slats can also be selected. The highest **positive contribution** to the **C_{LMAX}** is with **The** slats **from the retracted to the take-off position.** Slats increases the critical **angle of attack.**
An aeroplane has the following flap positions 0°, 15°, 30°, 45°; slats can also be selected. **Flaps from 30° to 45°** will most **adversely affect the C_L/C_D ratio (Flap down give more drag than lift, less C_{LMAX} and less (C_L/C_D ratio)**



Vortex generators transfer energy from the free airflow into the boundary layer.

When **vortex generators** are fitted they will normally be found **near the wing leading edge.**

The purpose of **vortex generators** is to **delay stall by reducing boundary layer separation.**

Means to Reduce the C_L - C_D Ratio

With extension of a **wing spoiler**, if the angle of attack remains constant **C_D is increased and C_L is decreased.**
When **spoilers** are used **as speed brakes at same angle of attack, C_D is increased and C_L is decreased.**
The result of **spoiler** surfaces deploying are **drag increases and lift decreases.**

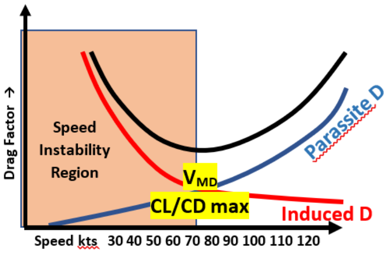
When **spoiler** are **extended in straight and level flight**, if the **speed and load factor** remain constant **C_D increases but C_L remains unaffected.**

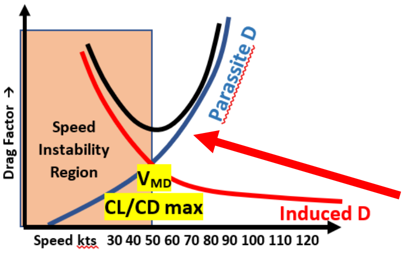
Speed brakes (all spoilers symmetrically) are devices used **to increase drag in order to maintain a steeper gradient of descent**. (**are used to descend rapidly, keep airplane down on landing**, thrust reverse is used to reduce speed on landing)
Spoiler surfaces on the top surface of a wing when operated symmetrically **may be used as speed brakes in flight.**

Wing **spoilers** are deflected **symmetrically** in flight in order to **decelerate the aeroplane and/or increase its rate of descent.** (All spoilers symmetrically it is simply to decelerate or increase descent rate. Speed brake mode. When turning, spoilers will move asymmetrically to prevent a yawing turn.)

When roll **spoilers are extended**, the part of the wing on which they are mounted **experiences a reduction in lift, which generates the desired rolling moment. In addition there is a local increase in drag, which suppresses adverse yaw. (rolling is annulled by the opposite spoiler)**

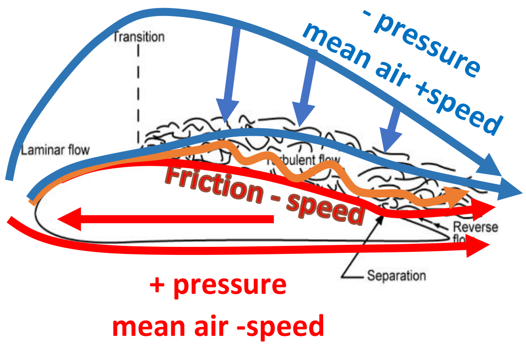
Spoiler extension **increases the stall speed, the minimum rate of descent and the minimum angle of descent.**





When **air brakes** are deployed **the minimum drag speed will reduce**. (Spoiler create more parasite drag so the parasite curve is steeper and moves to the left and minimum drag speed is reached at lower speed.)

The Boundary Layer



When considering the properties of a laminar and turbulent boundary layer **friction drag higher in turbulent.**
Friction drag is lower in the laminar layer.
(lower air at 0 speed on skin, the speed above is faster and is mixed with the lower speed below in a turbulent condition)
A turbulent boundary layer is between the transition and separation points.

Form drag is higher in **laminar flow. (form drag = Airflow impact on leading edge)**

A laminar boundary layer is thinner than a turbulent one.
A laminar boundary layer is thinner and has less friction drag than turbulent layer
A laminar boundary layer is a layer where no velocity components exist normal to the surface.
(Component = are two force that influence a vector. Normal=opposed)

A turbulent boundary layer produces more friction drag than a laminar one.

The turbulent boundary layer has more kinetic energy than the laminar boundary layer
The turbulent boundary layer separate less than the laminar boundary because lower pressure such the airflow to the wing. One important **advantage** a turbulent boundary layer **the turbulent boundary layer has less tendency to separate from the surface.**
Compared with a laminar boundary layer, a turbulent boundary layer is better able to resist a positive pressure gradient before it separates.

The **transition point** located on the wing is the point where **the boundary layer changes from laminar flow to turbulent flow.**

A laminar boundary layer turns into a turbulent one at the transition point.
Behind the transition point the mean speed and friction drag increase. (mean air increase due to Bernoulli theory)

Boundary layer has the greatest change in velocity close to the surface when considering velocity profile perpendicular to the flow

Aerodynamic Degradation

With icing during flight (crit) Angle of attack and controllability decrease, stall speed increase.

Ice, snow or frost formation Decrease critical angle of attack.

Effect of ice, snow or frost is decrease in lift and an increase in drag.

The largest ice build-up will occur, principally, on the frontal areas of the aircraft.

On the skin of an aircraft Skin Friction Drag is going to increase with increasing age
The drag due to an ageing airframe is identified as Skin Friction Drag
Effect of ageing of an aeroplane on the boundary layer is the transition point moves forward and a larger part is turbulent. Ageing drag is identified in parasite Drag

The effects of very heavy rain (tropical rain) on the aerodynamic characteristics of an aeroplane are decrease of C_{LMAX} and increase of drag.
The most important problem of ice is reduction in C_{LMAX}.

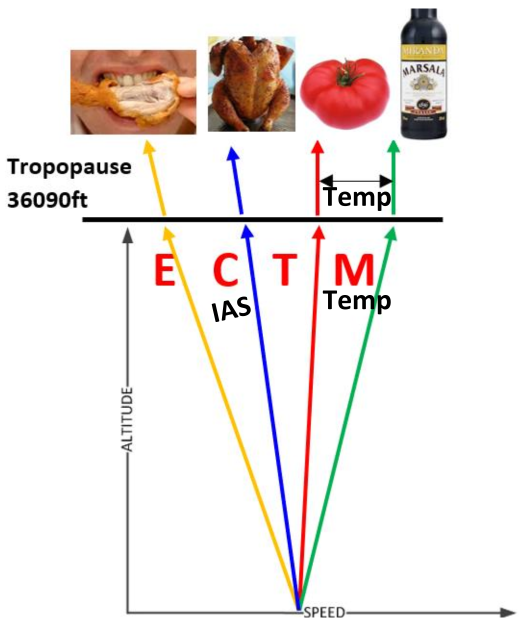
The aerodynamic effect of ice is most critical during the rotation of the take-off

If ice is present on the leading edge of the wings landing distance is increase by 40-50%

High Speed Aerodynamics

Speeds

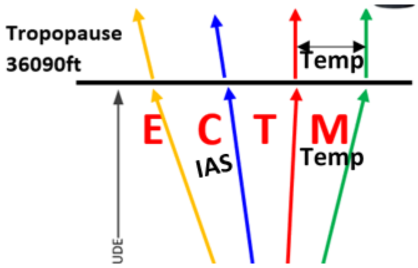
Eat Chicken Tomato Marsala



A **descent** **below** the tropopause at **constant Mach number** and aeroplane **mass, lift coefficient** **decreases**. (**TAS² increase, coefficient of lift decrease**, formula ever in balance)
A **climb below** the tropopause at **constant Mach number** and aeroplane **mass**, **the lift coefficient** **increases**. (**TAS² decrease, coefficient of lift increase**, formula ever in balance)

During a **descent** at a **constant Mach number the angle of attack will decrease**. (Tas is higher)
At a given Mach number, **air velocity increases with decreasing altitude.** (Air velocity = Tas)

Assuming ISA conditions and a **climb above the tropopause** at constant **Mach number** and aeroplane mass **lift coefficient increases.** (Tas remain the same above, but **density decrease** so coefficient of lift increase, formula in balance)



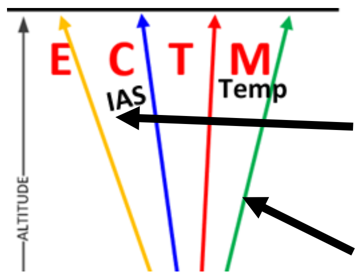
Assuming ISA conditions and a climb **above** the tropopause at **constant Mach number** and aeroplane mass, the **TAS remains constant.** **(Mach: TAS ÷ LSS(38.95 x √K)**

If **IAS is increased** when flying at **FL390**, **Mach number increase**

Mach number during a climb at **constant IAS** from sea level to 40 000 ft **Increases with increasing altitude.**

During a **descent** at a **constant Mach number the pitch angle will decrease.** (more density, more tas, less angle)

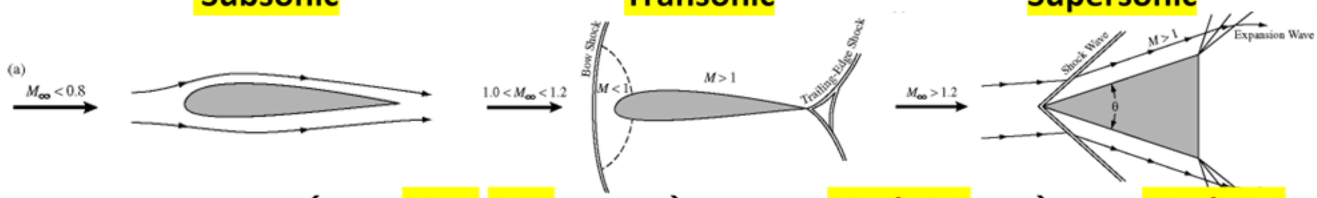
Compressibility effects depend on **Mach number**



An aircraft is **descending** at a **constant Mach number** and a constant weight. The operational speed that may be exceeded is
V_{MO} (descending, Vmo is the speed that increase more)
A jet transport aeroplane is in a straight **climb** at a **constant IAS** and constant weight. The operational limit that may be exceeded is
M_{MO} (climbing, Mmo is the speed that increase more)

V_{MO} can be **exceeded in a descent** at a **constant Mach** number because **V_{MO} is an IAS and descending at a constant Mach will result in an increase in TAS which will increase dynamic pressure.**

Subsonic **Transonic** **Supersonic**



up to ← **M_{CRIT} 0.75** → **Mach 1.3** → **Mach 5.0**

The **subsonic** speed range end at **M_{CRIT} 0.75**

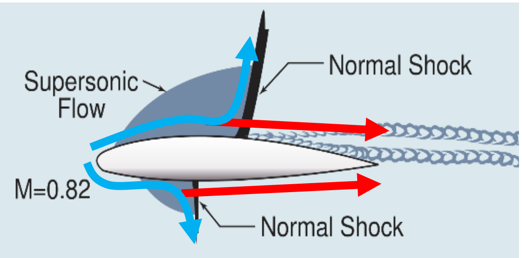
Transonic speed range is the regime of flight from the critical Mach number **up to approximately M = 1.3**

Transonic speed range is the region around Mach 1.

The speed range from approximately **M = 1.3** to approximately **M = 5** is called the **supersonic range.**

In **supersonic** flight aerofoil pressure distribution is **rectangular**

In **supersonic** speed **the airflow everywhere around the aeroplane is supersonic.**



In the **Transonic speed** range the aeroplane characteristics are strongly determined by the **Mach number.** (in **transonic** range, aeroplane have characteristic in order to **increase the Mach number** and **retard shock wave** and **drag**)

The Transonic speed range both subsonic and supersonic speeds exist in the flow around the aircraft.

At **Transonic speed** at which **both subsonic and supersonic local speeds occur**.

$$\mathbf{LSS = 38.95 \times \sqrt{K}} \qquad \mathbf{K = (c^\circ + 273)}$$

$$\mathbf{Mach = TAS \div LSS} \rightarrow \mathbf{LSS = Mach \div TAS}$$

The Mach number is the ratio between the **TAS of the aircraft and speed of sound of the undisturbed flow.**

The Mach number is 0.8 and the TAS is 400 kts. The local speed of sound is 500 kts (TAS ÷ Mach)

The Mach number is 0.8 and the TAS is 480 kts. The speed of sound is 600 kts (TAS ÷ Mach)

If CAS is 250 kts and TAS is 380 kts and OAT is -45°C, the Mach number is **0,65 M**

(LSS = 38.95 x √K K= (c°+273) Mach: TAS ÷ LSS

Using **mental navigation**, the **local speed of sound** may be found using **LSS = 644 + 1.2 TAT°c**

TAS or LSS are proportional to temperature, Mach is inversely proportional to temperature

Speed of sound increases with temperature increase. (Mach: TAS ÷ LSS(38.95 x √K)

The **air velocity (TAS)** for a given **Mach number increases when the air temperature increases**

For a **constant TAS** the **Mach number** of the airflow **increases when temperature decreases.**

(decrease of temp means climbing)

The **Mach number** corresponding to a given TAS will **be lower** if the **temperature increases.**

For a **constant flight** level and **IAS**, if the **OAT increases**, the **Mach number** will **remain constant**

The **speed of sound** is affected only by the **temperature** of the air

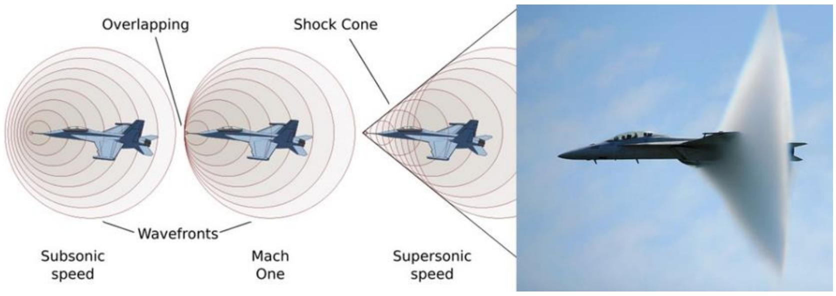
Speed of sound is **proportional** to the **square root** of the absolute **temperature** (unit Kelvin). Compare **Maximum Operating Mach number (M_{MO})** with **Maximum Operational Speed (V_{MO})**. To **fly as fast as possible** in **descent** from a **high cruising level**. **At first the airplane is limited by its M_{MO}, thereafter - below a certain altitude - by its V_{MO}.**

The **Mach number** of an airflow is determined by

The airspeed (Mach = TAS ÷ LSS)

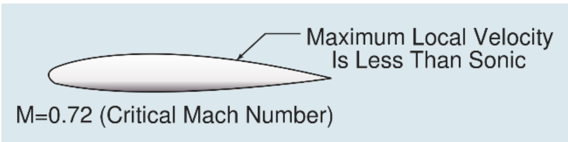
The air temperature (Mach = TAS ÷ (38.95 x √K))

Supersonic Shock Waves



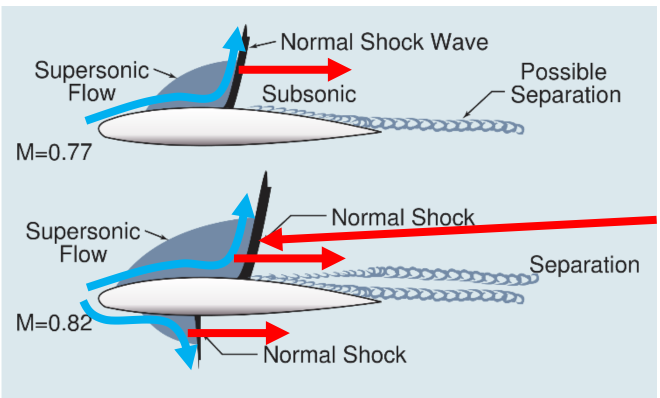
Every object moving in the air **produce a series of pressure waves in front of it** that travel at the speed of sound in all direction.

A sonic boom is the sound associated with the **shock waves created by an aircraft** traveling through the air faster than the speed of sound. This **shock wave** is caused by the **airflow** that **is not able to diverge** in so rapid velocity, it is pushed and rapidly separated a right angle by the leading edge on the wing shape. The air **it is compressed on the wings.**

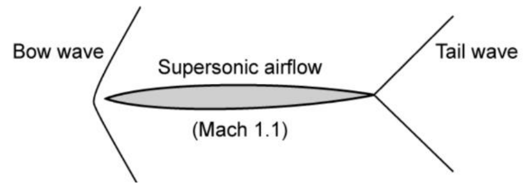


Every airplane not optimized for supersonic airflow have a different M_{CRIT} that limits the maximum speed . Transport airplane have usually a Mach Crit at 0.75.

Below M_{CRIT}, the airflow is subsonic around the wings. At Mach C_{RIT} the airflow is local sonic all on the airplane wings.



If speed is increased just above the M_{CRIT} the Airplane is much faster than the diverging airflow and cause a shock wave for impact of this compressed supersonic air in the upper part. The first shockwave start in the upper side of the wings where the local airflow faster (above Mach 1) is pushed up at right angle and is no more able to remain attached to the wing. The rest of the air is still in subsonic flow. If the airplane continue to increase the speed to well above M_{CRIT} the air around the wings becomes completely in supersonic speed, another shockwave start in the lower part of the wings.



If the speed is increased more, both shockwave move aft to the trailing edge and another bow shockwave is formed in front of the nose of the airplane.

The **bow wave** will first **appear** at a free stream Mach number **just above M = 1**.

Critical Mach number is the highest speed possible **without supersonic flow** over the wing

As the **Mach number increases** in straight and level flight, a shock wave on the upper surface of the wing will **move towards the trailing edge.** M↑ Shockwave →

The **critical Mach number** of an aeroplane is the Mach number **above which, locally, supersonic flow exists somewhere over the aeroplane.**
Critical Mach number is the free stream Mach number at which **local sonic flow first exists on any part of the aeroplane.** The **critical Mach number** of an aeroplane is the free stream Mach number at which for the first time, **somewhere** on the aeroplane, **local sonic flow** occurs

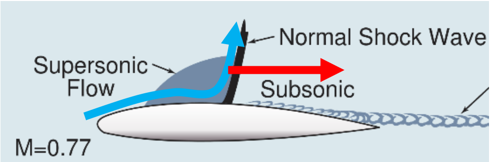
At speeds just **above** the **critical Mach number**, the **drag coefficient will start to increase.**
At speeds just **above** the **critical Mach number** the **L/D ratio** will **decrease.**
The **least energy loss** through a **normal shock** wave occurs when the local **Mach number** is **just above Mach 1. (just below Mach 1 or at exactly Mach 1 is not possible to have shockwave. Well above Mach 1 shockwave is very strong, just above Mach 1 shockwave give least energy loss).** For **minimum wave drag**, an **aircraft** should be operated at **Subsonic** speeds

The **drag divergence Mach number M_{CDR}**, also referred to as the "drag divergence speed", is the Mach number at which the aerodynamic drag on an Aerofoil or airframe begins to increase rapidly as the Mach number continues to increase. This increase can cause the drag coefficient to rise to more than 10 times its low speed value.
The **drag divergence Mach number** is the Mach number at which with increasing flow speed **the drag rises rapidly.**

The **drag divergence Mach number** of an aerofoil section is determined by
the angle of attack.
the profile of the aerofoil.

If the angle of attack is constant, the drag coefficient **will start to increase rapidly above the drag divergence Mach number. (not above M_{CRIT}, but above M_{CDR})**

A shock wave on a lift generating wing **will move slightly aft in front** of a downward deflecting aileron.



Above M_{CRIT}, a **shock wave** will appear first **near to the point of maximum wing thickness**.
Above M_{CRIT} the **first evidence** of a **shock** wave will appear at the **upper side of the wing.**

The first shock waves will occur **on the upper surface at the wing root.**
A **normal shock wave** is a discontinuity plane that is **always normal to the local flow**.~~surface~~
A **normal shock wave perpendicular** to **the local airflow**.
A **normal shock wave can occur at different points on the aeroplane in transonic flight.**
In **normal shock wave the airflow changes from supersonic to subsonic.**
In **shock wave the airflow changes direction**
Behind a normal shock wave on an aerofoil section the local **Mach number** is **less than 1.**

If a **shock wave appears** on the upper side of a wing or blade element, the **centre of pressure It will move aft towards the trailing edge.**

The **critical Mach number** of an aerofoil is the free stream Mach number at which **sonic speed (M=1) is first reached on the upper surface**.

Compared with an oblique shock wave **a normal shock wave** has **a higher loss in total pressure**.
Compared with an oblique shock wave **a normal shock wave** has a **higher compression.**

In case of **supersonic flow retarded by a normal shock wave**, a **high efficiency** (low loss in total pressure) **can be obtained**, if the **Mach number in front** of the shock wave is **small but still supersonic.**

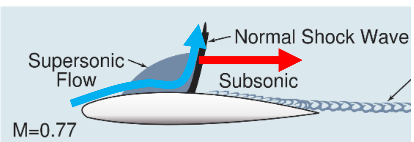
Air passes through a **normal shock wave. The static temperature increases. (like a wall)**

The **front of a shock wave move** across the **Earth's surface at the ground speed of the aeroplane.**

The effect of **aeroplane mass** on **shock wave intensity** at constant Mach number are **Increasing mass increases shock wave intensity.**

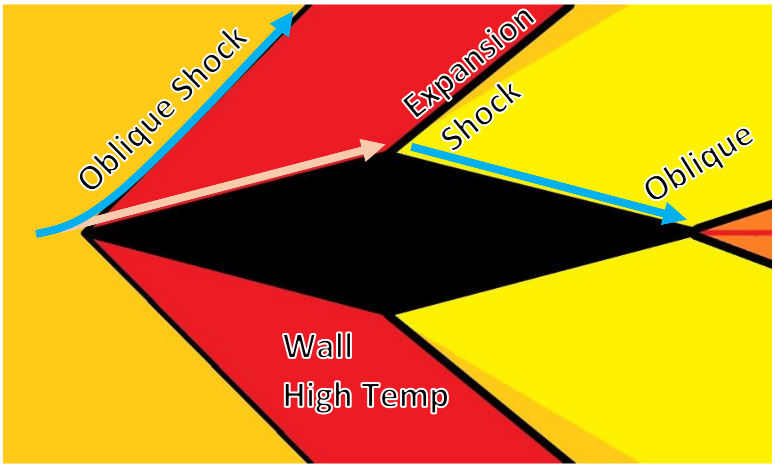
If IAS remains constant, the effect of **decreasing aeroplane mass** is that **M_{CRIT} increases.**
Whilst flying at a constant IAS and at n = 1, as the aeroplane **mass decreases**, the value of **M_{CRIT} increases.**
Influence of **decreasing aeroplane mass** on M_{CRIT} at constant IAS **M_{CRIT} increases as a result of flying at a smaller angle of attack**.

The **loss of total pressure in a shock wave** is due to the fact that **kinetic energy in the flow is changed into heat energy.**



The **velocity behind** a **normal** shock wave is **decreased to subsonic.**

To **increase the critical Mach number** of a conventional aerofoil section **its thickness to chord ratio should be reduced.**



The **velocity behind** an **oblique** shock wave is **decreased but still remains supersonic.**

Considering an oblique shock wave:
The Total pressure in front is higher
The Mach number in front is higher
The local speed of sound in front is lower
The static temperature in front is lower
The static pressure in front is lower
The density in front is lower

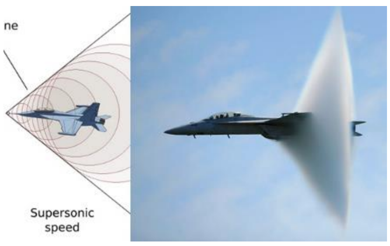
Considering an oblique shock wave:
The Total pressure behind is lower
The Mach number behind is lower
The local speed of sound behind is higher
The static temperature behind is higher
The static pressure behind is higher
The density behind is higher

Considering an expansion wave:
The Total pressure in front is lower
The Mach number in front is lower
The local speed of sound in front is higher
The static temperature in front is higher
The static pressure in front is higher
The density in front is higher

Considering an expansion wave:
The Total pressure behind is higher
The Mach number behind is higher
The local speed of sound behind is lower
The static temperature behind is lower
The static pressure behind is lower
The density in behind is lower

When the **air is passing through a shock wave**, the **density** will **increase. (air is compressed)**

When supersonic airflow **passes through an oblique shock wave**
static pressure increase
density increase **(air is pressed)**
local speed of sound increase



When an aircraft is flying at speeds **above Mach 1, pressure disturbances** from the **aircraft** will affect only the flow **within** the **Mach cone**. **(inside the cone)**
In **supersonic flight**, **all disturbances** produced by an aeroplane are **within the conical zone, depending on the Mach number.**

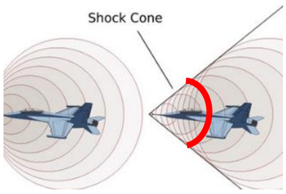
In supersonic flight, **any disturbance** around a body **affects** the flow only **within the Mach cone.**

The **sonic boom** of an aeroplane flying at supersonic speed is created by **shock waves around the aeroplane.**

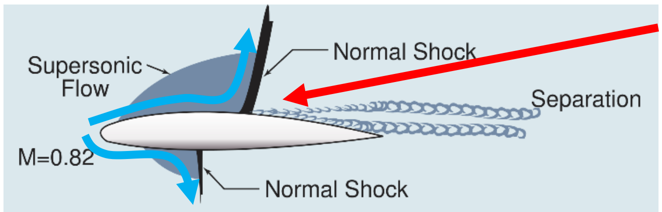
M = 1/sin mu

The value of the Mach number if the Mach angle equals 45° is **1.4**

The relation between the Mach angle (mu) and the corresponding Mach number is **sin mu = 1 / M**



If the **Mach number** in supersonic flight is **increased**, the Mach **cone angle will decrease**



Shock stall is a stall due to flow separation caused by a shock wave.
Shock stall is separation of the boundary layer behind the shock wave.
When **shock stall** occurs, lift will decrease because **flow separation occurs behind the shock wave.** **Shock induced** separation can occur **behind a strong normal shock wave, ranging from low to high angles of attack.** **Shock stall** occurs when the **lift coefficient**, as a **function of Mach number**, **reaches its maximum** value.
The **angle of attack** that have the **smallest** value is in **Shock stall.**
In the **transonic range** lift will decrease at the shock stall due to the **separation of the boundary layer at the shock waves.** **Shock induced separation** results in **decreasing lift.**

The additional **increase of drag at** Mach numbers **above the critical Mach number** is due to **drag wave**

Increasing aeroplane mass at **constant IAS, M_{CRIT} decreases as a result of flying at a greater angle of attack**

The **effect of a shock** wave **on control surface** hinge moment **will be rapid fluctuation of hinge moments, causing a high frequency "buzz".**

Shock waves cannot occur at speeds below M_{CRIT}.

On a typical transonic Aerofoil the **transonic rearward shift of the Centre of Pressure (CP)** occurs at about **Mach 0.89 to Mach 0.98**

M_{CRIT} is the free stream Mach number at which **somewhere about the airframe Mach 1 is reached locally.**

Deflecting a **control surface down** will cause **M_{CRIT} to decrease.** (Increased chamber with flap down accelerate supersonic flow, M_{CRIT} occur before/decrease.)

Effect of **exceeding M_{CRIT}** on the **stick force stability** of an aeroplane with swept-back wings without any form of stability augmentation is **a decrease, due to loss of lift in the wing root area.**

The consequences of **exceeding M_{CRIT}** in a swept-wing aeroplane may be **buffeting of the aeroplane and a tendency to pitch down.** (M-Tuck)

A **sharp increase** in **drag coefficient** of a given wing or blade section at a constant angle of attack is determined by the **Mach number of the airflow.** (exceeding M_{CRIT} create a huge increase of drag)

"**Tuck under**" is caused by **an aft** movement of the **centre of pressure (CP)** of the wing and **a decreasing** downwash angle at the location of the stabilizer (Cp move forward in subsonic, but move aft in supersonic speed)
"Tuck under" is caused by an aft movement of the centre of pressure of the wing.
"Tuck under" is caused by a reduction in the downwash angle at the location of the horizontal stabiliser. A contributing factor to " Tuck under " is a reduction in the downwash angle at the location of the horizontal stabiliser.
"**Tuck under**" is the **nose down pitching tendency as speed is increased in the transonic range**
"**Tuck under**" can occur **only above the critical Mach number.**

SUB-TRANS-SUPSC

The **Mach trim** system will prevent **tuck under.**
The **Mach trim** system is able to compensate for the **tuck under** effect by **decreasing the incidence of the trimmable tailplane.**
The **Mach trim** function minimize the adverse effects of **changes in the position of centre of pressure.**
The **effect of Mach trim** on stick forces for power operated controls **Is to maintain the required stick force gradient.**
In the event of **failure of the Mach trimmer the Mach number must be limited. (speed reduced)**
A **Mach trimmer corrects insufficient stick force stability at high Mach numbers.**
The **Mach trim** system will adjust **the stabilizer, depending on the Mach number.** ~~elevator~~
A Mach trimmer corrects the change in stick force stability of a swept wing aeroplane above a certain Mach number.

An aeroplane is flying through the **transonic** range whilst maintaining straight and level flight. As the **Mach number increases** the centre of pressure of the wing will move aft. This movement requires **a pitch up input of the stabilizer.**

The position of the **centre of pressure** on an aerofoil of an aeroplane cruising at **supersonic speed** when compared with that at subsonic speed is **further aft.**

An aeroplane should be equipped with a **Mach trimmer**, if **at transonic Mach numbers the aeroplane displays an unacceptable decrease in longitudinal stick force stability.**

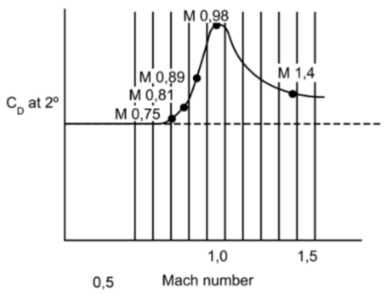
The movement of the **aerodynamic centre** of the wing when an aeroplane **accelerates through the transonic** range causes **an increase in static longitudinal stability.** (move to about mid chord (50% chord position). Pitch down moment means more stability)
When the speed over an aerofoil section increases **from subsonic to supersonic**, its **aerodynamic centre moves from approximately 25% to about 50% of the chord, move to mid chord.**

If a **symmetrical aerofoil** is accelerated from subsonic to supersonic speed the **centre of lift** will move **aft to the mid chord.** (50% chord position)

As the flight **Mach number increases** from subsonic to supersonic, the **centre of pressure** moves to **the mid chord position.**

When altitude increases, **the stall speed (IAS)** will **increase due to increasing compressibility effects as a result of increasing Mach number.**
The **increase in stall speed (IAS)** with increasing **altitude** is due to **the compressibility effects.**

In **transonic** flight the **ailerons will be less effective** than in subsonic flight because **aileron deflection only partly affects the pressure distribution around the wing.**



As an aircraft accelerates through the **transonic speed** range **the coefficient of drag increases then decreases.**

In the **transonic** range **C_{LMAX} will increase** and the 1g **stalling speed will decrease**

Remember shock wave: first increase drag, after passing the air drag decrease

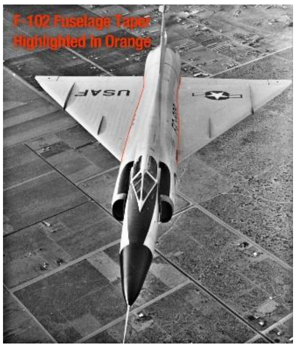
Means to Influence M_{CRIT}

A function of **vortex generators** in the transonic regime is to **reduce boundary layer separation drag when shock waves form.**
Vortex generators on the upper side of the wing **decrease wave drag (no M_{CRIT} but D_{RAG}).**
Vortex generators mounted on the upper wing surface will **decrease the shock wave induced separation.**

Two methods to **increase** the **critical Mach number** are **thin aerofoils and sweep back of the wing.**

M_{CRIT} is increased by **sweepback**, **thin aerofoils and area ruling.**

Sweep of a wing or rotorblade **delays** appearance of **shock waves** due to the **decrease of the velocity of the air, perpendicular to the leading edge** (decrease perpendicular air pushed at right angles, decrease waves)



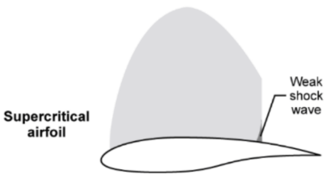
The application of the **area rule** on aeroplane design will **decrease** the **wave drag.**

Some aeroplanes have a "**waist" or "coke bottle**" contoured **fuselage**. This is done to **apply area rule.**

Increasing wing sweepback increases the drag divergence Mach number.
Decreasing wing sweepback decreases the drag divergence Mach number.
Increasing wing sweepback increases M_{CRIT}
Decreasing wing sweepback decreases M_{CRIT}.
Compared to straight wings of the same aerofoil section **sweep wings** have a **slower onset** of the **transonic drag** rise and a **higher C_D** in supersonic flight.
Compared to a straight wing of the same aerofoil section a wing with a **30º sweep-back** should theoretically have an **M_{CRIT} 1.154** times the M_{CRIT} of the straight wing. However, in practice the swept wing will gain **half** that increase.

A **sweepback wings** is one with the **least effective high lift devices**

The **critical Mach number** can be increased by **an increase of the sweepback of the wings.**
The **swept back wing** has the advantage of **higher critical Mach number.**

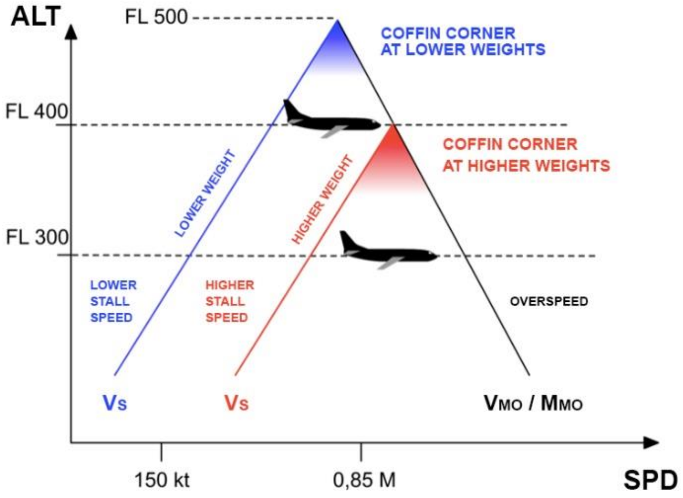


Typical shape characteristics of a supercritical aerofoil section are **a larger nose radius, flatter upper surface and negative as well as positive camber**. A **supercritical section** has **a flatter top surface.**
A **supercritical wing will develop no noticeable shock waves when flying just above M_{CRIT}**. One property of a **supercritical wing** aerofoil, compared to a conventional one is **it allows a wing of increased relative thickness to be used for approximately the same cruise Mach number.**

Large camber / thick aerofoil would result in the **lowest** value of **M_{CRIT}** **(tin aerofoil is better)**
Thick aerofoils and large angles of attack produce the **lowest M_{CRIT}** values

Reducing the thickness / **chord** ratio on a wing will
delay the onset of shock wave formation.
reduce the transonic variations in lift coefficient.
reduce the transonic variations in drag coefficient.

Buffet Onset

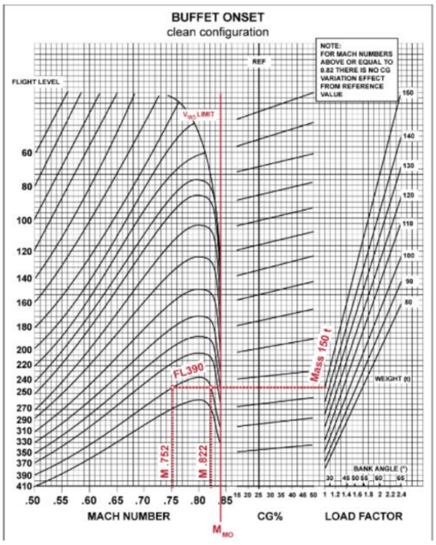


The critical speed where the **speed is too low and too high at the same time** is called **coffin corner**

The **speed range** between high and low speed **buffet increases during a descent at a constant IAS.**

A jet aeroplane is cruising at high altitude with a Mach number, that provides a buffet margin of 0.3g incremental. In order to **increase the buffet margin** to 0.4g incremental the pilot must **fly at a lower altitude and the same Mach number.**

Buffet onset, Aerodynamic ceiling. Higher and lower Stall speed



For **each altitude and mass** there are **2 limiting speeds** - **low and high speed buffet**. These 2 speeds limit the range of operating speeds at this altitude and mass. As the aeroplane climbs higher, this speeds range get narrower. At the maximum altitude there will be only **1 speed available**. **High speed buffet** is an aerodynamic limitation referred to as the "**coffin corner**".
In practice the pilot would never fly at aerodynamic ceiling (not to be confused with the service ceiling). In **aerodynamic ceiling** if you encounter a gust and your speed decreases, you **stall because of low speed condition**; if the gust causes a speed increase you **stall because of overspeed and shock waves form**.

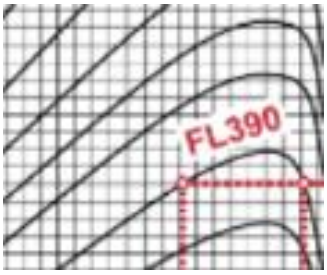
Mach buffet are **Transonic buffet** caused by the **separated turbulent boundary** layer striking the airframe with considerable force causing a high amplitude **vibration**, which physically shakes the whole aircraft.
Mach buffet occurs following **boundary layer separation** due to shock wave formation.
Mach buffet is a flight phenomena that can only happen at Mach numbers **above the critical Mach number**

The **high speed buffet** is induced by **boundary layer separation due to shock wave formation.**
The **high speed buffet** is induced by **interaction between shock wave motion and flow separation.**

From the **Buffet Onset Boundary chart** you obtain **the values of the Mach number at which low speed and Mach Buffet ~~shock stall~~ occur at different weights and altitudes.**

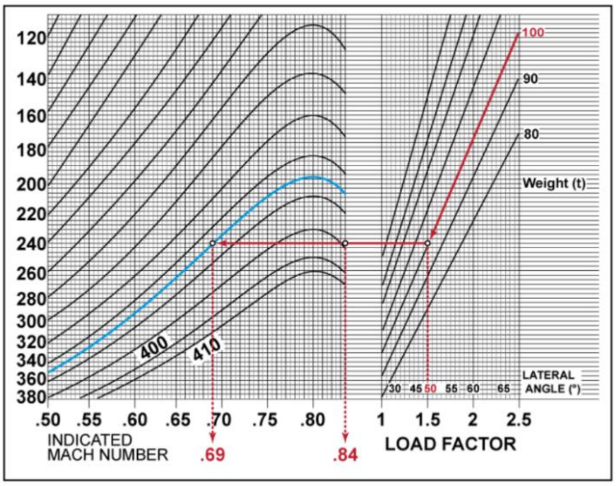
If a jet **aeroplane slowly accelerates** in level flight from its cruise speed in still air **at high altitude the type of buffet are Mach buffet.**

No, Is not acceptable that a transport aeroplane fly at a higher Mach number than the 'buffet-onset' Mach number in 1g flight



During a push-over manoeuvre the buffet free range increases.
When initiating a steady horizontal turn from steady straight level flight the buffet free range decreases.
When mass decreases the buffet free range increases.
When mass increases the buffet free range decreases.
When altitude increases the buffet free range decreases.
When altitude decreases the buffet free range increases.
When speed increases the buffet free range does not change. (Altitude and mass influence)

The significance of V_A for jet transport aeroplanes **is reduced at high cruising altitudes** because **buffet onset limitations normally become limiting.**

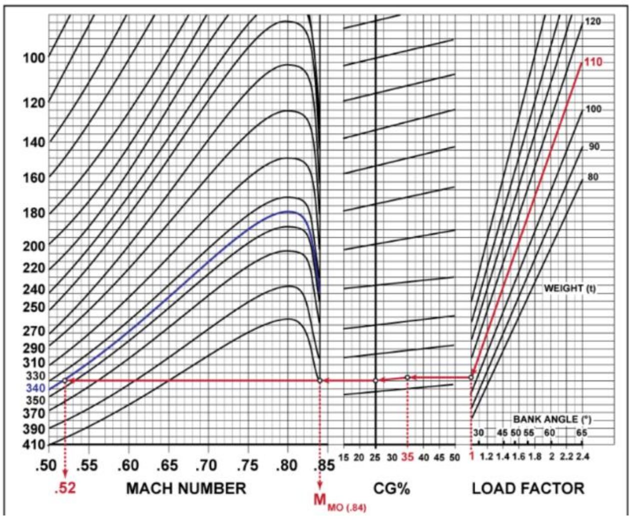


Buffet Onset Boundary Chart - Clean Configuration

A jet transport aeroplane weighing **100 tons** carries out a steady level **50° bank** turn at **FL350**. The buffet free speed range extends from **M = 0.69 to M > 0.84.**

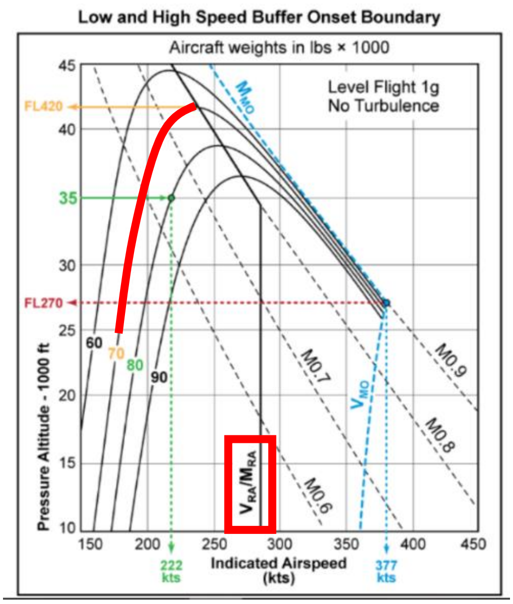
The maximum cruise altitude can be limited by a **1.3 g load factor** because when exceeding that altitude **turbulence** may induce **high speed** or **low speed buffet**.

A manoeuvre with a load factor of 1.3 will cause buffet onset.



Level flight: 1g
Cruising at: FL340
Aircraft mass: 110 000 kg
CG: 35 %
Low buffet onset speed: M 0.52
High buffet onset speed: M 0.84

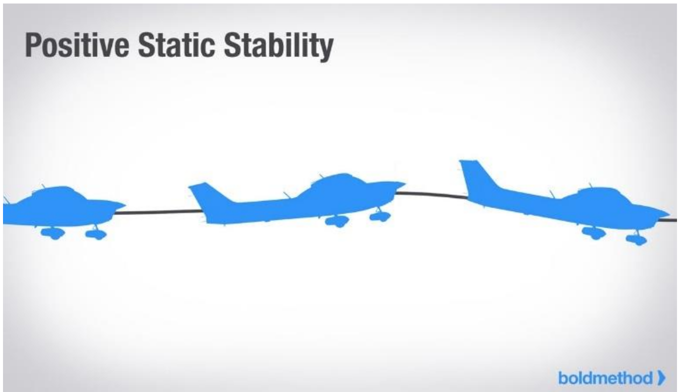
It is determined that at FL310 at a given mass buffet free flight is possible between M = 0.74 and M = 0.88. **If the aeroplane is suddenly pulled up** the **lower Mach number increases** and the **higher Mach number decreases.**

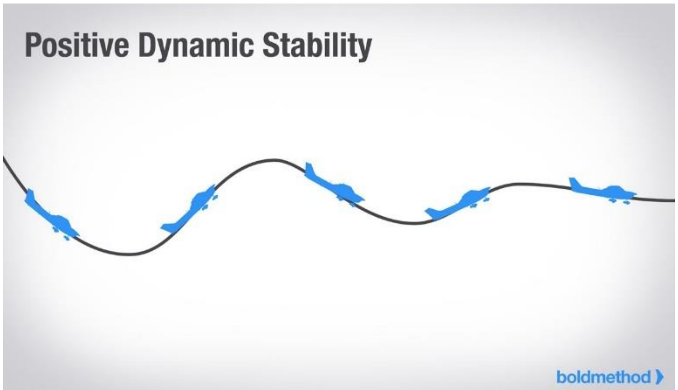


At an aircraft weight of **70 000 lbs** your **aerodynamic ceiling** in 1g level flight will be **FL420**

Stability

Static and Dynamic Stability





Static Stability is the initial tendency of an aircraft to return to its original position when it's disturbed.

Static Positive (Remain in initial attitude)

Static Neutral (Remain in new attitude when disturbance finish)

Static Negative (Continue to go further away after get a disturbance)

Dynamic Stability means how an airplane responds over long time to a disturbance. If airplane is statically unstable, cannot be dynamically stable, but if the airplane is statically stable it can be dynamic unstable.

Dynamic Positive (Remain in initial attitude)

Dynamic Neutral (Remain in new attitude when disturbance finish)

Dynamic Negative (Continue to go further away after get a disturbance)

Static stability means that **following a disturbance from the equilibrium condition, a force and/or moment is generated that tends to counter the effects of that disturbance.**

Positive static stability of an aeroplane means that following a disturbance from the equilibrium condition **the initial tendency is to return towards its equilibrium condition.**

An aeroplane that tends to **return to its pre-disturbed equilibrium** position after the disturbance has been **removed** is said to have **positive static stability. (if disturbance is not removed and return in equilibrium is dynamically stable)**

If an aircraft has **static** longitudinal **instability**, it **will be dynamically unstable.**
A **statically unstable** aeroplane is **never dynamically stable.**
Dynamic stability is possible only when the aeroplane is statically stable about the relevant axis.

A **statically stable** aeroplane **can show positive, neutral or negative dynamic longitudinal stability.**
An aeroplane that has **positive static stability** can be **dynamically stable, neutral or unstable.**

If an object is **statically unstable**, it will **move in the direction of the displacement. (diverge)**

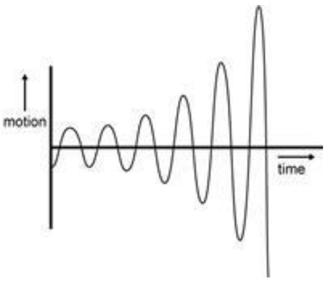
After encountering a **disturbance** an aeroplane **oscillates** about the lateral axis at a **constant amplitude.** The aeroplane is **statically unstable - dynamically neutral.**

As the **stability** of an aeroplane **increases its manoeuvrability decreases.**
As the **stability** of an aeroplane **decreases its manoeuvrability increases.**

If the sum of **moments in flight is not zero**, the aeroplane will **rotate** about the **centre of gravity.**

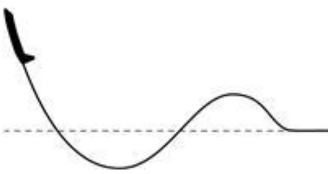
If the total **sum of moments** about one of its axis **is not zero**, an aeroplane would **experience an angular acceleration about that axis.**

For an aeroplane to possess **dynamic stability**, it needs **static stability and sufficient damping.**
One of the **requirements** for positive **dynamic stability** is **positive static stability**



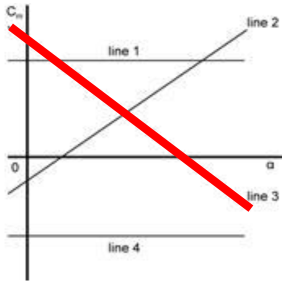
The aeroplane motion, schematically illustrated in the annex, is an example of a dynamically **unstable periodic motion.**

If a body is **dynamically unstable**, any oscillations would be **divergent**



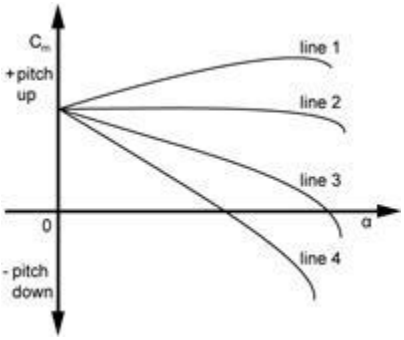
If an aircraft has **negative dynamic** and **positive static stability**, this will result in **divergent oscillations.**

Static and Dynamic Longitudinal Stability



Line 3 in pitching moment coefficient Cm versus angle of attack shows a **statically stable aeroplane**

A **static airplane** must have a **decreasing angle of attack**

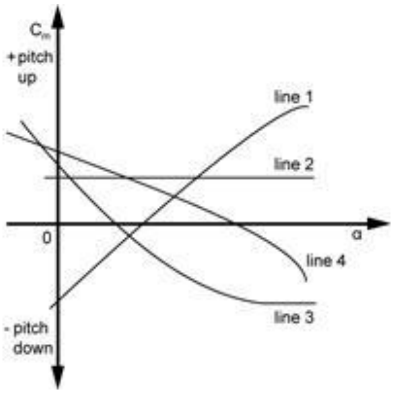


The **pitching moment** coefficient versus **angle of attack**
The CG position is further aft at line 1 when compared with line 4.

In its curved part at high angles of attack line 2 illustrates increasing static longitudinal stability.

CG located at the neutral point of a given aeroplane at **low and moderate angles of attack** is **line 2.**

The CG position is further forward at line 4 when compared with line 1.

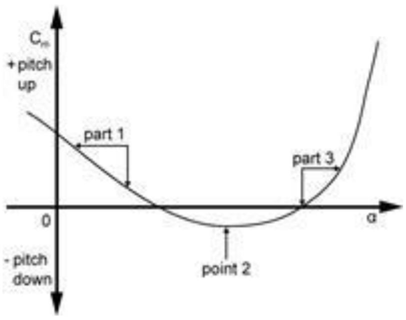


Regarding the pitching moment coefficient Cm versus angle of attack diagram
Line 1 shows an aeroplane with reducing static longitudinal instability at very high angles of attack.
Line 1 have static longitudinally instability at all angles of attack

Line 2 have neutral static longitudinal stability at all angles of attack

Line 3 shows an aeroplane with reducing static longitudinal stability at high angles of attack.
Line 3 shows an aeroplane with greater static longitudinal stability at low angles of attack than that shown in line 4.
Static longitudinal stability is greater at line 4 when compared with line 3 at low and moderate angles of attack.

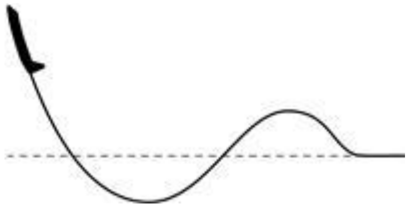
Line 4 shows an aeroplane with increasing static longitudinal stability at very high angles of attack. Line 4 is statically longitudinally stable at all angles of attack



The aeroplane exhibit **static** longitudinal **stability** in the part 1

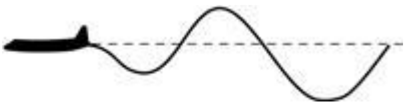
The aeroplane exhibit **neutral** static **longitudinal** stability on **point 2**

The aeroplane exhibit **static longitudinal instability on part 3**



Assuming no pilot input the motion of the aeroplane in the diagram shows **static longitudinal stability and dynamic longitudinal instability**.

Assuming no pilot input the motion of the aeroplane in the diagram shows neutral **dynamic longitudinal stability.**



Assuming no pilot input the motion of the aeroplane in the diagram shows **dynamic longitudinal stability.**

The value of the manoeuvre stability of an aeroplane is **150 N/g**. The **stick force required** to achieve a load **factor of 4** from steady level trimmed flight is **450 N.**

(load factor of 1 is 150, to reach 4 increase 3 times to 450)

When moving the **CG forward** the **stick force** per g will **increase**.

Every transport airplane have a manoeuvre **stability of 1g and the airplane can be forced to the limit manoeuvring of 2,5 g force.**

When the **stick force required** to achieve a **load factor of 2.5** from steady straight and level trimmed flight **is 225 N**, the value of the **manoeuvre stability (means 1g)** of that aeroplane is **150 N (2,5g-1g = 1,5. 225 / 1,5 = 150 N)**

When the **stick force required** to achieve a **load factor of 2.5** from steady straight and level trimmed flight is **375 N**, the value of the **manoeuvre stability (means 1g)** of that aeroplane is **250N (2,5g-1g = 1,5. 375 / 1,5 = 250 N)**

The **manoeuvre stability (means 1g)** of a large jet transport aeroplane is **280 N/g**. **Stick force** required, if the aeroplane is **pulled to the limit manoeuvring load factor (means 2,5g)** from a **trimmed horizontal** straight and steady flight (cruise configuration) is **420 N (if 280 N = 1g. limit 2,5g – 1g = 1,5g required. So 280 N x 1,5 = 420 N)**

If a stick force of **20 lbs** is required to pull **4g** from the position of trim, the **stick force gradient** is **6.6 lbs/g (4g-1g = 3g. 20 / 3 = 6,6 lbs)**

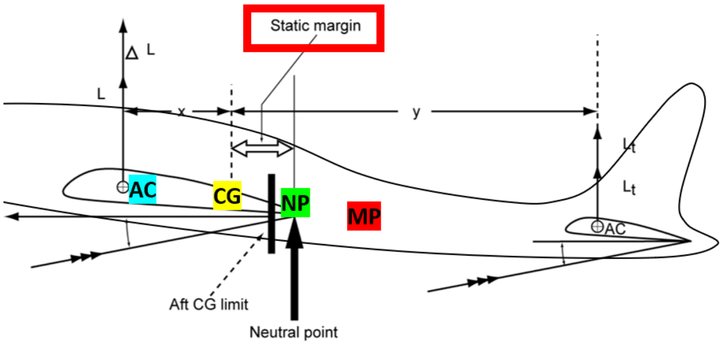
If the maximum pull force acceptable is **50 lbs** and the design limit load factor of the aircraft is **6g**, stick force/g that must be achieved to permit the aircraft to be manoeuvred to its **design load factor limit is 10 lbs (6 - 1 = 5 . 50 / 5 = 10)**

If the **CG is located beyond the forward** CG limit **This can lead to an unacceptably high value of manoeuvre stability. (is too negative stable that can be difficult to pitch up)**

The **neutral point** of an aeroplane is the point where **the aeroplane becomes longitudinally unstable when the CG is moved beyond it in an aft direction.**

For a **statically stable** aeroplane, the relationship between the neutral point and centre of gravity (CG) is such that the **neutral point** is located **aft of the CG.** **(NEUTRAL POINT EVER AFT)**

Position of the CG in relation to the neutral point when the CG is on the aft limit is **In front of the neutral point.**



If the aircraft is **properly loaded the CG**, the **neutral point** and the **manoeuvre point** will be in the order given, forward to aft:

CG, neutral point, manoeuvre point.

The "manoeuvre point" is the CG position where the **moment from the wing lift and the tail lift are equal during manoeuvring**, and this position is always behind the neutral point. "**Neutral point**" is the point along the longitudinal axis where the position of the CG causes **neutral stability** Consequently, an unstable aircraft shows a natural tendency to diverge from the trimmed flight attitude. Therefore, the **CG must be always located IN FRONT** of the **neutral point** during all phases of flight and the distance between the CG and the neutral point is referred to as the "**static margin**". The **distance** between the **CG Datum** and the **CG Neutral Point** in straight and level flight is called the **CG static Margin**

For a normal **stable aeroplane**, the **centre of gravity** is located **with a sufficient minimum margin ahead of the neutral point of the aeroplane.**

CG ahead of the **neutral point** ensures static longitudinal **stability**

The **magnitude of the stick force** required to pitch, for an aircraft with manual controls, is determined by **the distance the CG is forward of the neutral point.**

Longitudinal static stability is created by the fact that the **centre of gravity is located in front of the neutral point of the aeroplane.**

The **CG** of an aeroplane is in a **fixed position** forward of the neutral point. **Speed changes** cause a departure from the **trimmed position**. **An increase of 10 kts from the trimmed position at low speed has more effect on the stick force than an increase of 10 kts from the trimmed position at high speed.**

If an airplane exhibits poor longitudinal stability in flight, **to increase the stability** we can **Increase the horizontal stabilizer surface area.**

Positive static longitudinal stability means that a **nose down moment occurs after encountering an up-gust.**

Longitudinal stability is directly influenced by **centre of gravity position.**

The **aft CG limit** can be determined by the **minimum acceptable static longitudinal stability.**

An **aft CG** shift **increases longitudinal manoeuvrability.**
An **aft CG** shift **decreases static longitudinal stability.**

The most **aft CG location** may be limited by:
insufficient stick force stability.
excessive in-flight manoeuvrability.

The contribution of the **wing** to the **static longitudinal stability** of an aeroplane **depends on CG location relative to the wing aerodynamic centre.**

The **manoeuvrability** of an aeroplane is **best** when the **CG is on the aft CG limit.**

The **forward CG** limit is determined primarily by **minimum control response**. The **aft CG** limit is determined primarily by **decreasing stability**.

A **forward CG** shift **increases static longitudinal stability.**
Forward movement of the CG will reduce control response and increase stability.
A **forward CG** shift **decreases** longitudinal **manoeuvrability**.
Forward movement of the CG will **increase stability** and **reduce controllability.**
When the **CG is close to the forward** limit **very high stick forces are required in pitch because the aircraft is very stable.**

The most **forward CG location** may be limited by:
insufficient flare capability.
insufficient in-flight manoeuvrability.

With the **CG on the aft limit** the **control forces** required to pitch the aircraft would be **less than with a forward CG.**

An in-flight move of the **CG** of an aircraft further **aft** will **reduce the longitudinal stability.**
If **CG** moves behind the **aft limit** there is **Insufficient manoeuvre stability.**
A **CG** location **beyond** the **aft limit** leads to **an unacceptable low value of the manoeuvre stability.**

The **centre of gravity** moving **aft** will **increase the elevator up effectiveness.**

The **max aft** position of the **centre of gravity** is amongst others **limited** by the **minimum value of the stick force per g.**

The effect of a **Centre of Gravity (CG)** shift to a more **aft location** is **the static longitudinal stability is reduced and the required control deflection is smaller.**

Effect of an **aft shift of the centre of gravity** on static longitudinal stability and the required control deflection for a given pitch change **Static longitudinal stability reduces and the required control deflection reduces.**

Given an aeroplane in **steady, straight and level flight at low speed**, the **lowest value of wing lift is required** at **CG aft and high thrust.**
Given an aeroplane in steady, straight and level flight at low speed, the **highest value of wing lift is required** at **forward CG and low thrust.**

An aeroplane, with a **CG location behind the centre of pressure** of the wing can only maintain a **straight and level flight** when the horizontal tail loading is **upwards**.

The horizontal tailplane provides the greatest positive contribution to static longitudinal stability

For an aircraft in steady level flight, **if the tail plane** is producing a **download,** the **CP of the wing** must be **aft of the CG.**

Positive camber of an aerofoil **not affect** static longitudinal stability **no effect, because camber of the aerofoil produces a constant pitch down moment coefficient, independent of angle of attack.**

The effect of the **wing downwash** on the static longitudinal stability of an aeroplane is **negative.**

The purpose of the **horizontal stabilizer** is to give the aeroplane **sufficient longitudinal stability**.

A longitudinal dynamic oscillation can take two forms. One of these, the "**long period oscillation**", involves slow changes in **altitude and speed.**

When an aeroplane has **zero static longitudinal stability**, the **pitching moment coefficient** "Cm" versus angle of attack line **is horizontal.**

TRIM HAS NO EFFECT ON STABILITY

With a **downward** adjustment of a **trim tab** in the longitudinal control system **the stick position stability remains constant.**
The **effect** of elevator **trim tab adjustment** on the static longitudinal stability of an aeroplane **No effect**.
Upward deflection of a **trim tab** in the longitudinal control results in **the stick position stability remaining constant.**

During landing of a low-winged jet aeroplane, the **maximum elevator up deflection** is normally required when the **flaps** are **fully down and the CG is fully forward.**

The stick force per G must have both an upper and lower limit in order to assure acceptable control characteristics.

A **pitch up** could be caused by **forward movement of the centre of pressure.**

A **phugoid** refers to **dynamic longitudinal stability.**

Damping is the property that **slows down the rate or diminishes the amplitude of vibrations or cycles.**

Damping of the phugoid is normally very weak.
The short period oscillation should always be heavily damped.

Long period (phugoid) oscillations are characterised by **long period of damping.**

A short period oscillation should always be heavily dampened. (Short period oscillation are more difficult to be controlled by the pilot)
If the airspeed increases and decreases during longitudinal **phugoid** oscillations, the aircraft **can be easily controlled by the pilot.**

During a **phugoid** the speed **varies significantly**, whereas during a **short period oscillation** it **remains** approximately **constant**.

During a phugoid altitude varies significantly, but during a short period oscillation it remains approximately constant.
During a **short period** oscillation, the altitude **remains approximately constant, whereas during a phugoid it varies significantly.**

The **short period mode** is an **oscillation about the lateral axis.**
The **phugoid** motion is a **long term oscillation** around the **lateral axis.**

If the aircraft has a **nose up pitch displacement**, the **effective angle** of attack of the **tail plane changes and causes the tail plane to apply a nose down moment.**
When an aeroplane with the **centre of gravity forward** of the centre of pressure, the **vertical load on the tailplane** will be **downwards**.

The **air loads** on the horizontal **tailplane (tail load)** of an aeroplane in straight and level cruise flight are generally directed **downwards and will reduce in magnitude as the CG moves aft.**

At constant EAS, as **height increases Damping in all axes is reduced. (less density, less damping)**

A pilot **recognise** static stick force **stability** when **to maintain a speed below the trim speed requires a pull force.**

A pilot **recognise** static stick force **stability** when **To maintain a speed above the trim speed requires a push force.**

A **negative contribution** to the static longitudinal stability of conventional jet transport aeroplanes is provided by **the fuselage.** **(fuselage is heavy, nose down = negative contribution)**

The **stick force gradient** is **the force required to change the load factor of the aircraft a given amount.**

A high limit load factor enables the manufacturer to design for a lower stick force per g.

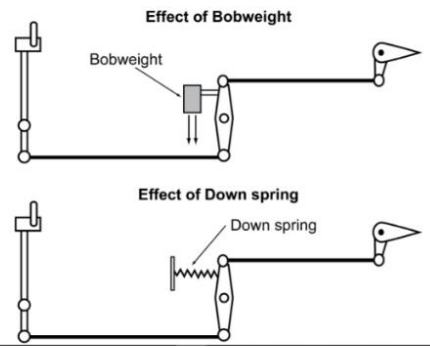
Stick force per g is **dependent on CG location.**

The **stick force** stability and the **manoeuvre stability** are positively **affected** by **A forward CG movement in case of the stick force stability and a forward CG movement in case of the manoeuvre stability**

A requirement for positive static longitudinal stability of an aeroplane is, that the neutral point is behind the centre of gravity.
A wing with positive camber provides a positive contribution to static longitudinal stability, when the centre of gravity of the aeroplane is in front of the aerodynamic centre of the wing.

An aeroplane exhibits **static longitudinal stability**, if, when the **angle of attack increases the change in total aeroplane lift acts aft of the CG. (CoP aft of CoG=CoG forward)**

The **aerodynamic contribution** to the static longitudinal stability of the **nacelles of aft fuselage mounted engines** is **always positive.**
aerodynamic contribution means disregard thrust.



A bob weight and a down spring have the same effect on the stick force stability.

Static Directional Stability

Directional stability is the stability around the **normal axis. (vertical)**

With **lateral wind** the **nose point** ever to **the wind direction**

The contribution to the **static directional stability** of a **straight wing with high aspect ratio** and without dihedral **is always negligible.**
The contribution of **swept back wings** to **static directional stability is positive**
The effect of a **wing sweepback** on static directional stability is a **stabilizing effect.**

Regarding a sideslip, with the relative **airflow coming from the left**, on an aeroplane that exhibits both directional and lateral stability (remember, wind push on tail)
The initial tendency of the nose of the aeroplane is to move to the left
The initial tendency of the right wing is to move down.

Regarding a sideslip, with the relative **airflow coming from the right**, on an aeroplane that exhibits both directional and lateral stability (remember, wind push on tail)
The initial tendency of the nose of the aeroplane is to move to the right
The initial tendency of the left wing is to move down.

An aeroplane has static directional stability. In a **sideslip to the right**, initially the **nose** of the aeroplane tends to **move to the right.**

An aeroplane has static directional stability if, when in a **sideslip** with the relative **airflow coming from the left**, initially the **nose of the aeroplane tends to yaw left.**

Static **directional** stability is the **tendency of an aeroplane to recover from a skid without control input from the pilot.**

A **dorsal fin** provides a positive contribution to **static directional** stability
The purpose of a **dorsal fin** is to **maintain static directional stability at large sideslip angles**. Static **directional stability** is mainly provided by **the fin**
Increasing the **size of the fin increases the directional stability.**
A dorsal fin increases the contribution of the vertical tail plane to the static directional stability, in particular at large angles of sideslip.

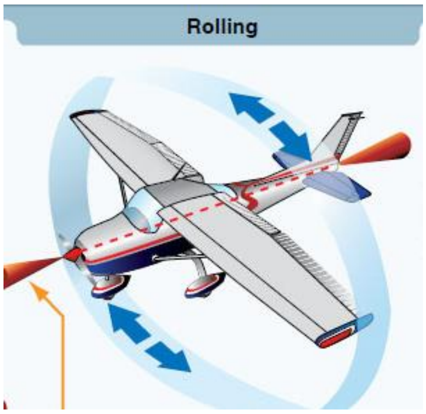
A dorsal and a ventral fin both have a positive effect on static directional stability, Not ~~lateral~~

The effect of a **ventral fin** on the static stability of an aeroplane is as follows:
longitudinal – **no effect**
lateral - **negative**
directional - **positive**

An aeroplane's **sideslip angle** is defined as the angle between the **speed vector and the plane of symmetry.**

An aircraft demonstrates **positive static directional stability**. This relate to the interaction between the aircraft's **yawing moment coefficient C_n** and its sideslip angle (ß)
C_n is positive and the aircraft experiences a positive sideslip angle.

Static Lateral Stability



You improve static **lateral** stability by
High wing.
Large and high vertical fin.
Dihedral
Sweepback

You reduce static **lateral** stability by
Anhedral.
Forward sweep.
Low wing (less stable than high wing)
Ventral fin. (less stable than dorsal)

Sweepback of a wing positively influences:
static lateral stability, roll
~~static longitudinal stability,~~ pitch up/down
~~dynamic longitudinal stability.~~ pitch up/down

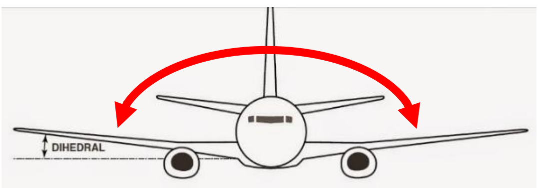
Sweep back provides a positive contribution to static lateral stability.

Static **lateral stability** will be **increased** by the use of a **high**, rather than low, **wing mounting**.
Static **lateral stability** will be **decreased** by **the use of a low, rather than high, wing mounting.**

Lateral static **stability** is determined by **aircraft response to sideslip. (stability is a response)**

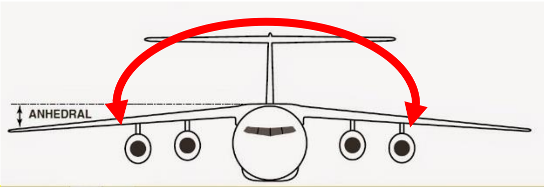
The effect on static lateral stability of an aeroplane with a **high wing** as compared with a low wing is **a positive dihedral effect.**

Positive static **lateral stability** is the **tendency** of an aeroplane to **roll to the left in the case of a sideslip (with the aeroplane nose pointing to the left of the incoming flow).**



The primary purpose of **dihedral** is to **increase static lateral stability.**
An increase in geometric **dihedral** in a steady sideslip condition at constant speed would **increase the required lateral control force.** (is more stable, control force are higher to roll)
To **hold a given sideslip** angle and airspeed, increased **dihedral** would **increase the stick force.**

The effective dihedral of an aeroplane component means the contribution of that component to the static lateral stability. Wing **dihedral contributes to static lateral stability.**



Static **lateral stability** will be increased by **reducing wing anhedral. Anhedral** increase static lateral **manoeuvrability** and decrease lateral stability. Static **lateral stability** will be **decreased** by **increasing wing anhedral.**
Anhedral decreases the static lateral **stability** of an aeroplane.

A **slideslip to the right** is created by **Right wing down and left rudder.**
A **slideslip** to the **left** is created by **Left wing down and right rudder.**

Static lateral stability will be decreased by **reducing wing sweepback.**
Static lateral stability will be increased by **increasing wing sweepback.**

Sweepback of a wing provides a **positive dihedral effect.**

One advantage of mounting the **horizontal tailplane** on top of the **vertical fin** is **to improve the aerodynamic efficiency of the vertical fin.**

Static **lateral stability** should **not be too large**, because **too much aileron deflection would be required in a crosswind landing.**
Excessive static lateral stability is an **undesirable** characteristic for a transport aeroplane because **it would impose excessive demands on roll control during a sideslip.**

When flaps are extended Lateral stability is decreased. (roll better, more lift)

For an aircraft with **neutral static lateral stability**, following a wing drop **the wing would remain in its displaced position.**

To increase the static **lateral stability By increasing the aspect ratio of the vertical stabiliser, whilst maintaining a constant area.**

For a **sideslip** condition at constant speed and sideslip angle, where the geometric **dihedral** of an aeroplane **is increased the required lateral control force increases.**

An aeroplane's **bank angle** is defined as the angle between its **lateral axis and the horizontal plane.**

Dynamic Lateral/Directional Stability

Moments or motions that interact in a **Dutch roll** are **Rolling and yawing.**

An example of a **combined lateral and directional** periodic **motion** is a **Dutch roll.**

Dutch roll can happen at Mach numbers **below the critical Mach number**

Yaw damper suppresses the tendency to **Dutch roll**
Dutch roll is corrected by a **yaw damper**

Dutch roll = + more lateral stable (+Pitch)
Spiral = + more directional stable (+Yaw)

An aeroplane is sensitive to **Dutch roll** when **static lateral stability is much more pronounced than static directional stability.**

An **increased static lateral stability** will increase the tendency to **Dutch roll**

Recommended action following **failure of the yaw damper** of a jet aeroplane is to **Reduce altitude and Mach number.**

If the **static lateral stability** of an aeroplane is **increased**, whilst its static directional stability remains constant **its sensitivity to dutch roll increases.**

An aircraft's tendency to **Dutch roll** may be **reduced** by **increasing** the **anhedral angle** of the wings.

An example of a **combined lateral and directional aperiodic motion** is a **spiral dive.**

Sensitivity for **spiral dive** will occur when **the static directional stability is positive and the static lateral stability is relatively weak. (spiral roll instability)**

An aeroplane with an excessive static directional stability in relation to its static lateral stability, will be prone to spiral dive (spiral roll instability)

Static **lateral stability** should **not be too small** because **the aeroplane would show too strong a tendency to spiral dive.**

An aircraft is placed in a level balanced turn and the controls released. It is **spirally unstable** if **the bank steadily increases.**

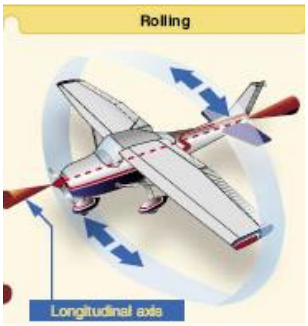
With **swept back wings**. With increasing altitude and **constant IAS** the **static lateral stability** will **increase** and the **dynamic lateral-directional stability decreases.**
With a **swept wing aircraft**, with an increase in altitude, **Static lateral stability remains the same, dynamic lateral-directional stability decreases.**

Control

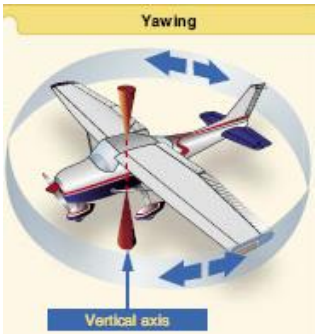
General

Aeroplane **manoeuvrability increases** for a given control surface deflection when **IAS increases.**
Aeroplane **manoeuvrability decreases** for a given control surface deflection when **IAS decreases.**

The **axes of an aircraft** by definition must all pass through the **centre of gravity.**



Rotation about the longitudinal axis of an aeroplane can be achieved by **aileron deflection** and/or **rudder deflection**.
Rolling is the **rotation** of the aeroplane about the **longitudinal axis.**
Rotation around the **longitudinal axis** is called **rolling**
Lateral control is achieved by **the ailerons, resulting in a motion around the longitudinal axis.**
The function of **ailerons** is to **rotate** the aeroplane about the **longitudinal axis.**



The **normal axis** of an aircraft is **an axis passing through the CG, perpendicular to the longitudinal axis.**
Directional control is achieved by **the rudder, resulting in a motion around the normal axis.**
Rotation around the **normal axis** is called **yawing**
The pilot uses the **rudder** to provide control around the **normal axis.**
Rudder controls **yaw**

The **lateral axis** is also sometimes referred to as the **pitch axis.**
The **pitch angle** is defined as the **angle between** the **longitudinal axis and the horizontal plane.**
Rotation about the **lateral axis** is called **pitching**
Pitch is movement around the **lateral axis.**
The **elevators** control the aircraft around **the lateral axis.**
The **elevator** is the primary control surface for control about the **lateral axis** and is operated by a forward or backward movement of the control wheel or stick.

To pitch an aeroplane **nose up**, the pilot applies backpressure on the control column and moves the elevator relatively **upwards, increasing the negative camber of the tailplane and elevator.**

When the **control column is moved back** and to the left **the left aileron moves up and the elevators move up.**
When the **control column is moved forward** and to the **right the elevators move down, the right aileron moves up and the left aileron moves down.**

Given
theta = pitch angle
gamma = flight path angle

alpha = angle of attack

theta = gamma + alpha. (pitch angle = flight path angle + angle o attack)

Pitch (Longitudinal) Control

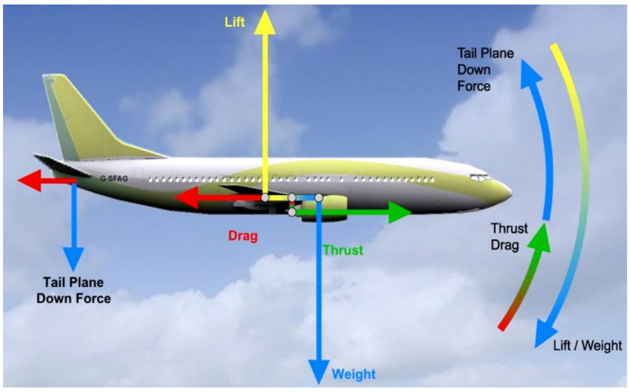
Center of Gravity (CG) position is predominantly used to **determine the correct setting** (position) of the **trimmable horizontal stabilizer** for take-off

When **ice is present** on the **stabilizer**, deflection of **flaps** may cause **the stabilizer to stall and a vertical dive.**

An aircraft is approaching to **land with its CG at the FWD limit**. Compared to a situation with CG at the AFT limit, the **control forces** in pitch during the **flare** will be **higher** and **V_{REF}** would be **higher** .

When the **landing gear is lowered**, given that the **CG does not move** longitudinally, to maintain level flight, the **download on the tail plane** must **increase**

On an aircraft with an **all-moving tail plane**, **backward** movement of the control wheel in flight causes **a decrease in tail plane incidence and a nose up pitch.**

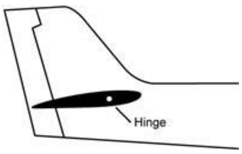


A jet transport aeroplane exhibits **pitch up when thrust is suddenly increased** from an equilibrium condition, because the **thrust line is below the CG.** (low engine)

If an **increase in power** make the **nose to rise**, this is the result of the **line of thrust being below the CG.**

Low speed pitch-up can be caused by a **significant thrust increase with podded engines located beneath a low-mounted wing.**

Advantage of a **variable incidence tailplane** over a fixed incidence tailplane with elevator and trim tab is **Less trim drag and maximum elevator authority retained.**



Horizontal control surface shown in the figure is **All-flying tail.**

An **advantage of engines mounted on the rear fuselage** over those mounted in wing pods **Longitudinal trim is less affected by changes in thrust.**

An **advantage** of locating the **engines at the rear of the fuselage**, in comparison to a location beneath the wing, is **less influence on longitudinal control of thrust changes. (aft engine have smaller asymmetric problems due to inboard position of engine)**

On a jet aeroplane (**engines mounted below** the low wing) if the **thrust** is suddenly **increased** the **elevator** must be deflected **downward**.

For a given elevator deflection, aeroplane **longitudinal manoeuvrability increases** when **the CG moves aft.**
For a given elevator deflection, aeroplane longitudinal **manoeuvrability decreases** when **the CG moves forward.** If the **centre of gravity** is moved to **forward the manoeuvrability will decrease.**

The **elevator deflection** required for a given manoeuvre will be **larger at low IAS when compared to high IAS.**
The **elevator deflection** required for a given manoeuvre will be **smaller at high IAS when compared to low IAS.**

The **elevator deflection** required for a given manoeuvre will be **smaller for a aft CG position when compared to an forward position.**
The **elevator deflection** required for a given manoeuvre will be **larger for a forward CG position when compared to an aft position.**

When the **CG position is moved forward**, the required **elevator deflection** to achieve an **increase in load factor** will be **larger.**
When the **CG position is moved forward**, the **elevator deflection** to achieve a **decrease in load factor** will be **smaller.**

The forward CG limit is mainly determined by the amount of pitch control available from the elevator.

When the **CG moves** from the aft limit **to the forward limit** and the pilot wants to make a change in pitch **manoeuvrability decreases due to static longitudinal stability.**

A **conventional stabilizer** on a stable aeroplane in a normal cruise condition:
contributes to the total lift of the aeroplane.
may stall before the wing, in icing conditions, with large flap settings, unless adequate design and/or operational precautions are taken.
is necessary to balance the total pitch moment of the aeroplane.

Yaw (Directional) Control

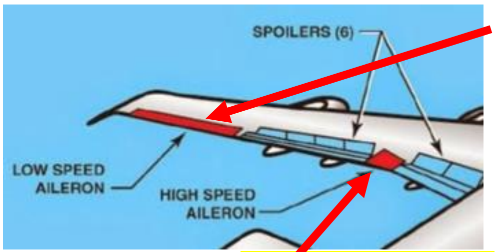
Full rudder deflection is limited on CAT transport aeroplanes, **A full rudder deflection could cause an excessive load on the structure of the aeroplane.**

If the **right rudder pedal** is pushed forward **the rudder moves to the right and the aircraft yaws to the right.**

When the rudder pedals are moved to cause a yaw to the left **the left pedal is moved forward and the rudder moves to the left.**

The **vertical fin** of an aircraft is a symmetrical aerofoil **It could stall if the fin angle of attack is too great.**

Roll (Lateral) Control



The **outboard ailerons** (if fitted) are **not active** in **cruise.**

Outboard ailerons (if present) are normally used **in low speed flight only.**

The purpose of **inboard ailerons to reduce wing twist at high speed.**

The **outboard ailerons** (if present) are **de-activated** with **Flaps (and slats) retracted or speed above a certain value.** The **outboard ailerons** are typically used only **when the flaps are extended**.

Aileron deflection causes a **rotation** around the **longitudinal axis** by **changing the wing camber and the two wings therefore produce different lift values resulting in a moment about the longitudinal axis.**

The **primary roll controls** on a conventional aeroplane are **the ailerons.**

A jet aeroplane equipped with inboard and outboard ailerons is cruising at its **normal cruise Mach number.** In this case **only the inboard ailerons are active.**

Adverse yaw during a turn entry is caused by **decreased induced drag on the lowered wing and increased induced drag on the raised wing.**

When a turn is initiated, **adverse yaw** is **the tendency of an aeroplane to yaw in the opposite direction of turn mainly due to the difference in induced drag on each wing.**

One method to **compensate adverse yaw** is a **differential aileron.**
Adverse yaw is **counteracted** by the use of **differential ailerons**
Adverse yaw is compensated during entry into and roll out from a turn by **differential aileron deflection.**

Correct example of **differential aileron** deflection to initiate a **left turn is Left aileron up 5° / right aileron down 2°.**

In a turn, **differential ailerons reduce the drag on the up-going wing.**

Differential aileron deflection equalises the **drag** of the right and left aileron.

On an aircraft with a **differential aileron** control system, when the **control wheel** is turned to the **right the left aileron moves down and the right aileron moves up through a greater angle.**

An aeroplane fitted with **differential ailerons** is initiating a level **turn to the left the left aileron moves up more than the right aileron moves down.**

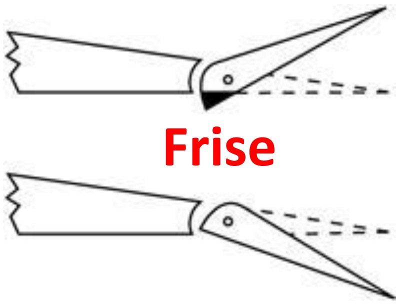
In a differential aileron control system the control surfaces have a larger upward than downward maximum deflection.

An aeroplane uses **rudder aileron cross-coupling** coupling to **prevent adverse yaw**. If the aeroplane is rolled to the right **The left aileron is deflected down, the right aileron is deflected up and the rudder is deflected to the right automatically. (rudder goes in direction of flight)**
An aeroplane uses **rudder aileron cross-coupling** coupling to prevent adverse yaw. If the aeroplane is **rolled to the left the left aileron is deflected up, the right aileron is deflected down and the rudder is deflected to the left automatically.**

When **rolling out of a steep banked turn**, what causes the lowered aileron to create more drag than when rolling into the turn is **the angle of attack of the wing being raised is greater as the rollout is started.**

Form drag is the component of drag that increases most when an **aileron is deflected upwards**

If a turbulent **gust causes** an aeroplane to **roll the up-going wing experiences an increase in angle of attack.**



Frise

When an aircraft is **rolled to the left**, adverse yaw will be reduced by **a Frise aileron being effective on the left wing.**
Frise ailerons have an **asymmetrical** leading edge that **protrudes below** the lower surface of the wing, causing high drag.
The **aileron** on which the **leading edge protrudes** below the wing when the aileron is raised, but not above it when the aileron is lowered is **a Frise aileron.**

For a given **IAS and angle of aileron** deflection, **increasing altitude** will **increase the rate of roll.**
(fixed IAS = Increase in Tas at higher altitude. Rate of turn is not reduced at high speed, but you must roll more to have the same rate of turn, roll rate increase)

Spoilers mounted on the wing upper surface can be used to **assist the ailerons.**

Spoilers are operated **asymmetrically to provide roll control.**

Spoilers installed on a typical large transport aeroplane are **upper wing surface devices and their deflection can be symmetrical or asymmetrical.**

Flaperons are controls, which combine the function of **flaps and ailerons.**

An aeroplane have **spoilers** and **both inboard and outboard ailerons**. Roll control during cruise is provided by **inboard ailerons and roll spoilers.** In this case **outboard ailerons are locked out.**

Aircraft is **descending** in straight flight with the **spoilers deployed**. If a pilot initiates a descending turn to the left the **spoilers move up on the down-going wing and down on the up-going wing.**

During initiation of a turn with **speed brakes extended**, the roll spoiler function induces a spoiler deflection **downward on the up-going wing and upward on the down-going wing.**

In an aircraft fitted with **spoilers** for lateral control, and not deployed as speed brakes, a **roll** to the **right** is initiated by **right spoiler extended, left spoiler retracted.**

In a **steep turn to the left**, when using flight **spoilers** for roll assistance and at the same time for reducing the airspeed **the right aileron will descend, the left one will ascend, the right spoiler will retract and the left one will extend.**

On an aircraft on which the **ailerons** are **assisted by spoilers** to give lateral control, if the **control wheel** is turned to the **right the right aileron moves up, right spoiler up, left spoiler remains retracted, left aileron down.**

Roll/Yaw Interaction

If the **nose** of an aeroplane **yaws to the left**, this causes **a roll to the left.**

The **secondary effects of rudder and aileron** deflection **rudder: roll; aileron: yaw**

Yaw is **followed by roll** because the **yawing motion generated by rudder deflection causes a speed increase of the outer wing, which increases the lift on that wing so that the aeroplane starts to roll in the same direction as the yaw.**

Left rudder pedal input will cause a **left yaw about the vertical axis and left roll about the longitudinal axis.**

Means to Reduce Control Forces

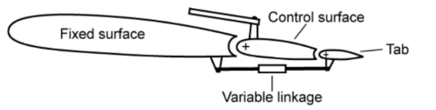
Artificial feel is required **with fully powered flight controls.**
The **primary input** for an **artificial feel** system is **IAS.**

The **aerodynamic force** on a control surface **increases as speed increases.**
The inputs to the **Q feel** unit are from **pitot and static pressures**.

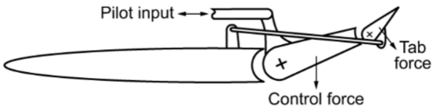
To **reduce the stick force** required, the **control surface** may be **aerodynamic balanced.**
The purpose of "**aerodynamic balance**" of a flying control is to **reduce the load required to move the control.**

Stick forces, provided by an **elevator feel system**, depend on **elevator deflection**, **dynamic pressure.**

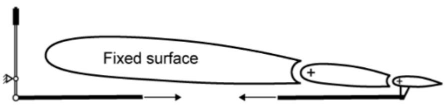
When **power assisted controls** are used for pitch control **a part of the aerodynamic forces is still felt on the column.**



The tab in the figure represents **a balance tab that also functions as a trim tab.**
Balance tabs move in the opposite direction to control surfaces and the same direction as trim tabs. A **balance tab** is **a form of aerodynamic balance.**



An **anti-balance tab** moves in the **same direction as the control surface and increases** control **effectiveness.**
Anti-balance tab is a devices used to provide **aerodynamic balance**



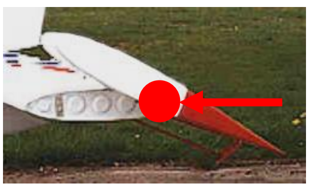
The tab in the figure represents a **servo tab.** A **servo tab moves opposite** of the **control surface** to which **it is hinged.**

Concerning **primary control surface** controlled by a **servo tab the position is undetermined during taxiing, in particular with tailwind.**

In a **servo tab** operated control system, movement of the tab is **always in the opposite direction to the control surface.**
A **servo tab** on the **rudder** moves **when the rudder pedals are moved. (pilots direct connected)**

An aeroplane has a **servo tab controlled elevator**. If the **elevator jams during flight Pitch control reverses direction.**
Servo tab Is most **commonly** used In case of **manual reversion** of fully powered **flight controls**

For an aircraft fitted with servo tab operated controls, if **external locks** are **fitted** to the main **control surfaces they will prevent movement of the control surfaces but not the control wheel or the servo tabs.**



An **aileron control surface** is provided with an **inset hinge to provide aerodynamic balance.** On an aileron with an **inset hinge**, the **leading edge** of the **control surface may protrude above or below the wing surface** when the control is moved. This is to **give assistance to the pilot to move the control.**

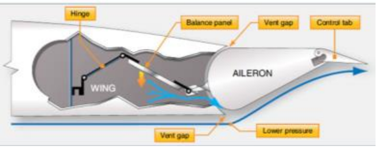
A **horn balance** in a control system has the purpose **to decrease stick forces.**
A **graduated horn** balance **prevents over balance resulting from excess balance at high speed.**

If the control **surface hinge is placed too far back** from the control surface leading edge control surface **CP may move ahead of the hinge and cause overbalance.**

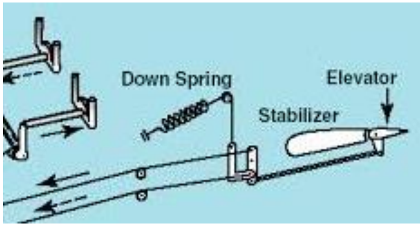
In general, **control forces** are reduced by **a horn balance, servo tab and spring tab.**
Horn balance, balance tab, servo tab can be used to **reduce control forces**
Servo tab - horn balance - spring tab decrease manoeuvring **stick forces (NO TRIM)**

Examples of aerodynamic balancing of control surfaces are **servo tab, spring tab, seal between the wing trailing edge and the leading edge of control surface.**
Examples of aerodynamic balancing of control surfaces are **seal between wing's trailing edge and leading edge of a control surface, horn balance.**

The fundamental **difference** between a **trim tab** and a **servo tab** is that **The purpose of a trim tab is to reduce continuous stick force to zero, a servo tab only reduces stick force.**



Aerodynamic balance can be obtained by **an internal balance**



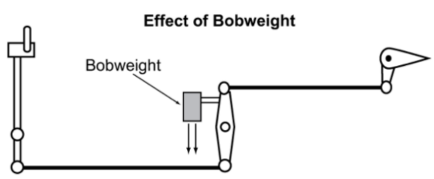
The spring tab is a modification of the servo tab such that the movement is proportional to the applied stick force.
A spring tab at high IAS it behaves like a servo tab.
Some airplanes have **spring tabs** to provide **reduction in the pilots effort to move the controls against high air loads.**

Mass Balance

Mass-balancing of control surfaces is used to **prevent flutter of control surfaces.**

On fully hydraulic powered flight controls there is no need for trim tabs.

When flutter damping of control surfaces is obtained by **mass balancing**, these weights will be located with respect to the hinge of the control surface **in front of the hinge. (increase stability)**



If an aeroplane exhibits **insufficient stick force per g**, this problem can be resolved by installing a **bobweight in the control system which pulls the stick forwards.**

Trimming

An aeroplane with a forward CG requires the stabiliser leading edge to be lower than for one with an aft CG in the same trimmed condition.

A **tab fitted to zero the loads on the pilot's** control column during flight is known as **a trim tab.**

In order **to zero the load** if a **rearward force on the control column** is required to maintain level flight **the elevator trim tab must be moved down.**

In general **jet transport aeroplanes** with **power assisted flight controls** are fitted with an **adjustable stabiliser** instead of trim tabs on the elevator. This is because **an adjustable stabiliser is a more powerful means to generate the tail loads required for these kind of aeroplanes.**

After an **aeroplane** has been **trimmed the stick position stability will be unchanged.**

The **reason** for having a **trim system** on power assisted flying controls is to **enable the stick force to be reduced to zero.**

The purpose of a **trim tab** (device) is to **reduce or to cancel control forces.**

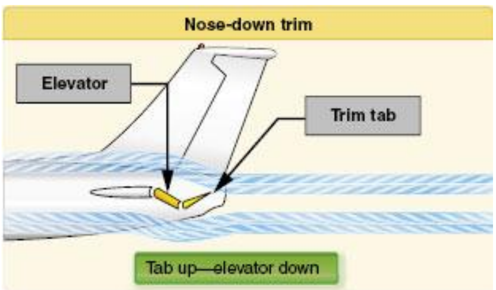
The trim tab **reduces hinge moment and control surface efficiency.**
The **position of the elevator** in relation to the **trimmable horizontal stabiliser** of an aeroplane with fully hydraulically operated flight controls **that is in trim** is **Elevator deflection is zero.**
(zero deflection in order to not create drag)

A large modern aircraft employs a **variable incidence tailplane**, **trim changes** are made by **changing the angle of incidence of the entire tailplane.**

When the **elevator** control surface is moved an **adjustable trim tab Remains fixed for all positions.**
(trim tab move only when pilot move trim wheel.)

In the unlikely event of an **aircraft becoming airborne with the elevator control lock** engaged, operation of an **elevator trim tab** control in the normal direction to counteract a nose up tendency would **result in an increase of the nose up tendency. (When aileron is jammed the trim act in opposite direction and increase the problem if used normally)**

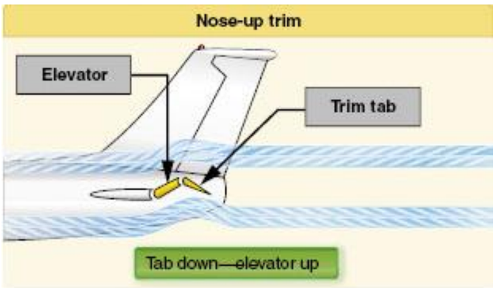
An elevator trim is less suitable for aeroplanes with a large CG range.



If the elevator **trim tab** is **deflected up**, the cockpit trim indicator shows **nose down.**

In straight flight, as **speed is increased**, whilst **trimming** to keep the stick force zero the **elevator is deflected further downwards** and the **trim tab further upwards.**

In straight flight, as **speed is reduced**, whilst **trimming** to keep the **stick force zero the elevator is deflected further upwards and the trim tab further downwards**.



If the elevator **trim tab** is **deflected down**, the cockpit trim indicator shows **nose up.**

If the **elevator trim tab** has been moved fully to the aircraft **nose up** position, the available aircraft **nose up pitch authority** will be **reduced.**

If the **control column is pushed forward**, a **trim tab** on the elevator **will not move relative to the elevator unless the trim wheel is moved.**

Deflecting the **elevator up,** when the **trim tab is in neutral**, will cause the tab to **remain in line with the elevator**

Comparing an **elevator trim** system with a **stabiliser trim** system
an elevator trim is more sensitive to flutter.
a stabiliser trim is less sensitive to flutter.

Comparing the **differences** between **a horizontal trimmable stabiliser** and an **elevator trim tab**
A stabiliser trim is a more powerful means of trimming.
A stabiliser trim is more suitable to cope with the large trim changes generated by the high lift devices on most jet transport aeroplanes.
The effects of a stabiliser trim runaway are more serious.
A horizontal trimmable stabiliser enables a larger CG range.
A trim tab is less suitable for jet transport aeroplanes because of their large speed range.
A jammed trim tab causes less control difficulty.
A trim tab runaway causes less control difficulty.
When trimmed for zero elevator stick force an elevator trim tab causes more drag.

A stabiliser trim is more suitable to cope with the large trim changes generated by the high lift devices on most jet transport aeroplanes.

The most **important factor** determining the required **position** of the **Trimmable Horizontal Stabiliser (THS)** for take-off is the **position of the aeroplane's centre of gravity.**
Aeroplane takes off with the CG at the aft limit and the **trimmable horizontal stabiliser (THS)** is at **nose down** position for take-off **rotation will be normal using the normal rotation technique.**
Aeroplane takes off with the CG at the forward limit and the **trimmable horizontal stabiliser (THS)** is at **nose up** position for take-off **rotation will be normal using the normal rotation technique.**
Aeroplane takes off with the CG at the aft limit and the **trimmable horizontal stabiliser (THS)** is at **nose up** position for take-off **early nose wheel raising will take place.**
Aeroplane takes off with the CG at the forward limit and the **trimmable horizontal stabiliser (THS)** is at **nose down** position for take-off **rotation will require a higher than normal stick force.**

About a **trimmable horizontal stabiliser A trimmed aeroplane with an aft CG requires the stabiliser leading edge to be higher than in the case of a forward CG in the same condition.**

Consider two elevator control systems, the **neutral position of the control column** after trimming for a **speed decrease**
fitted with a trim tab, Control column moves aft,
fitted with fully powered hydraulic controls and an adjustable horizontal stabiliser, **Control column does not change.**
for a **speed increase**
fitted with a trim tab, Control column moves forward,
fitted with fully powered hydraulic controls and an adjustable horizontal stabiliser, **Control column does not change.**

An aircraft is equipped with an "**all flying tailplane**" which has a combined **anti-balance** and **trimming tab.** The top of the **trim wheel is moved forward**. **The tab moves up, so that more effort is required when the pilot attempts to move the control column to the rear.**

One **advantage** of a **movable-stabiliser** system compared with an **elevator trim system** is that **it is a more powerful means of trimming.**

The **position of the elevator** in relation to the **trimmable horizontal stabilizer** of a power assisted aeroplane **the position depends on speed, the position of slats and flaps and the position of the centre of gravity.**

To perform a **landing** with the **stabilizer jammed** in the **cruise** flight **position** you **choose a higher landing speed than normal and/or use a lower flap setting for landing.**
The effect on **landing speed** when a trimmable horizontal **stabiliser jams** at high IAS is **In most cases, a higher than normal landing speed is required.**

A stabiliser trim is able to compensate larger changes in pitching moments.

To **counteract** a right **wing low tendency**, a **fixed tab** on the **left aileron** would **be moved down causing the left aileron to come up.**
When an **aileron trim** control in the cockpit is moved to **correct** a tendency to fly **left wing low**, an **aileron trim tab** on the **left aileron** will move **up which causes the left aileron to move down and the right aileron to move up.**

The primary purpose of equipping the **power assisted flying control systems** with **trim capability** is to **bring the control forces to zero in steady flight.**

Limitations
Operating limitations

V_{LE} is defined as the **maximum landing gear extended speed.**

V_{NE} is defined as **never exceed speed.**

M_{MO} can be **exceeded** in a **climb** at a **constant IAS** because **maintaining a constant IAS requires an increase in TAS.**

Wing flutter is **prevented by locating mass in front of the torsion axis of the wing.**

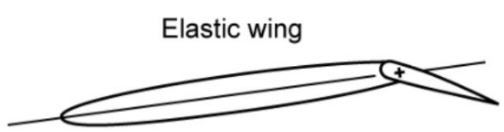
Wing mounted engines extending ahead of the wing contribute to wing flutter suppression.
Excessive free play or backlash reduces the speed at which control surface flutter occurs.
Resistance to flutter increases with increasing wing stiffness.
The risk of flutter increases as IAS increases.
If flutter occurs, IAS should be reduced.
Aero-elastic coupling affects flutter characteristics.

If **climbing at V_{MO}, it is possible to exceed M_{MO}.**

V_{MO} is a CS-25 aeroplane speed that may not be deliberately exceeded in any regime of flight (climb, cruise, or descent), unless a higher speed is authorised for flight test or pilot training operations.

Considering a high speed transport jet aeroplane, If **flaps** are selected to the **fully extended** position at a **speed greater than V_{FE} flap movement will typically be prevented by the flap load relief system.**

For most jet transport aeroplanes, the maximum operating limit speed, **V_{MO} is replaced by M_{MO} at higher altitudes.**



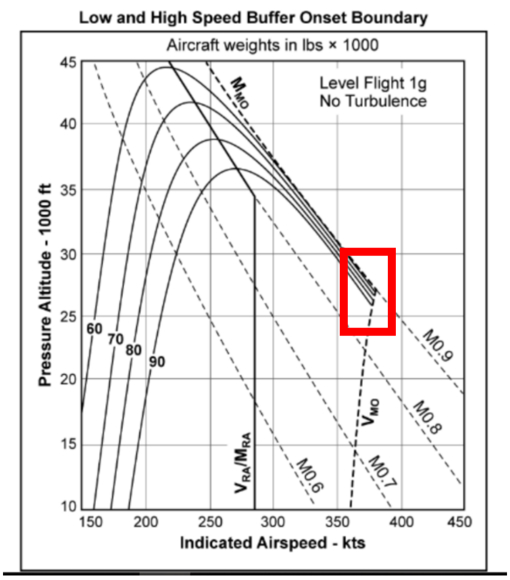
Aileron reversal at high C_L results from the **down-going aileron increasing the semi-span angle of attack beyond the critical.**
Aileron reversal may be caused by **the wing twisting and reducing incidence when the aileron is lowered.**

Aileron reversal may be caused by **twisting of the wing above reversal speed.**

Control surface flutter Is a destructive vibration that must be damped out within the flight envelope.
Control surface flutter can be **eliminated** by **mass balancing of the control surface.**
Flutter of control surfaces is **a divergent oscillatory motion of a control surface caused by the interaction of aerodynamic forces, inertia forces and the stiffness of the structure.**
Control **surface flutter** is **a rapid oscillation of the control surface in flight.**

Aileron flutter can be caused by <u>**cyclic deformations**</u> **generated by aerodynamic, inertial and elastic loads on the wing.**

Flutter sensitivity of an aeroplane wing **is reduced** by **locating the engine ahead of the torsional axis of the wing.**
Wing **flutter** may be caused by a **combination of bending and torsion of the wing structure.**
Wing **flutter** can be **prevented** by **ensuring that the wing CG is ahead of its torsional axis.**



In a **high speed descent at M_{MO}** you will **reach V_{MO}** at **FL270**

Manoeuvring Envelope

V_A decrease by half of the mass decrease

An aircraft is **flown at 20% below** its normal **mass**. Because of this, **V_A** will be <u>**10% lower.**</u>
When the aeroplane's **mass decreases by 19% the V_A is 10% reduced.**
When the aeroplane's **mass decreases by 10% the V_A is 5% lower.**

An aircraft has a **mass of 60 000 kg** and a limiting positive **load factor of 2.5 g**. **V_A** is calculated as the EAS at which full positive elevator deflection will give the limiting load factor at the stall, and is **237 kts.** If the aircraft **mass is reduced to 40 000 kg** by fuel burn, the new V_A is **194 kts** (reduced mass 33%, **speed reduce by half** of that number 17%)

Positive **manoeuvring limit load factor** for a large transport aeroplane with **flap extended** is **2.0**
The **limit load factor** of a large transport aeroplane is **2.5**
The **negative manoeuvring limit load factor** for a **large jet** transport aeroplane is **-1 g**
The **negative manoeuvring limit load factor** for a <u>**normal**</u> category aircraft is <u>**-1.52 g**</u>
The **manoeuvring limit load factor** for a **light aeroplane** (EASA CS-23) in the <u>**normal**</u> category is <u>**3.8**</u>
The **manoeuvring limit load factor** for a **light aeroplane** in the **utility** category is **4.4**

Manoeuvring limit load factor determines **V_A**
The **V_A** is the **maximum speed** at which **maximum** elevator **deflection** up **is allowed.**

If an aircraft is flying at a **speed above V_A a full elevator deflection could cause a structural damage.**

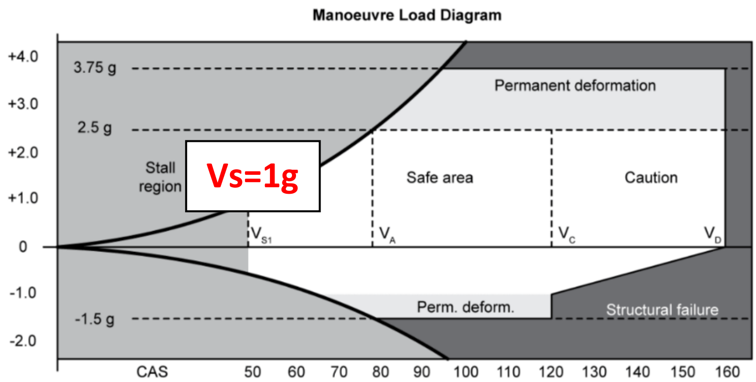
The **aeroplane structure** when flying at a speed just **exceeding V_A It may suffer permanent deformation if the elevator is fully deflected upwards.**

When **determining V_A load factor limit should be taken in account**

For an aeroplane with one **fixed value of V_A, V_A is the speed at which the aeroplane stalls at the manoeuvring limit load factor at M_{TOW}.**

If the "**ultimate load factor**" is exceeded may happen **Structural failure.**

The **manoeuvring speed V_A**, expressed as indicated airspeed, of a transport aeroplane **depends on aeroplane mass and pressure altitude.**



The **stall speed** line in the **manoeuvring load diagram** runs through a point where the **speed = V_S, load factor = +1.**

The **stall speed** lines in the manoeuvring load diagram originate from a point where the **speed = 0, load factor = 0.**

The **stall speed** line in the manoeuvring load diagram runs through a point where the **speed = V_A, load factor = limit load factor.**

Assuming no compressibility effects $V_A \geq V_S \times \sqrt{n}$

The relationship between the stall speed V_S and V_A (EAS) for a **large transport aeroplane** can be expressed in the following formula **$V_A \geq V_S \times \sqrt{2,5}$**

V_C is a speed which is chosen by the designer and which is used to assess the strength requirements in cruise.

V_{MO} should not be greater than V_C.

The **load factor** is **greater than 1** (one) **when lift is greater than weight.**

Gust Envelope

LOAD FACTOR = LIFT / WEIGHT

Gust load factor

Proportional - Wing area / slope lift X AoA / EAS

Inversely Proportional - wing load / Altitude / Mass

Load factor is **increased** by **upward gusts.**

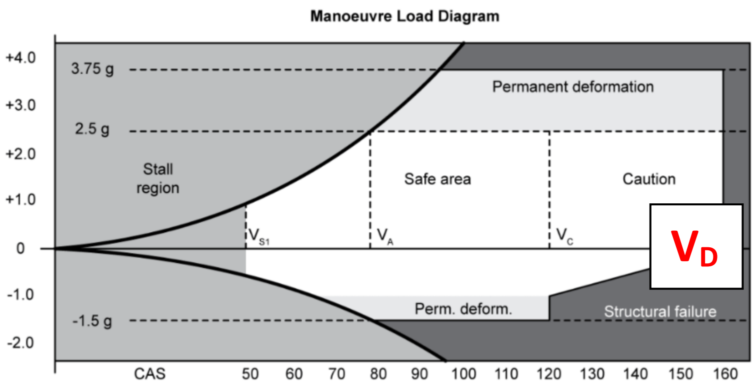
All other relevant factors being constant a load factor increase by a Vertical gusts.

When the EAS increases/decreases, the gust load factor increases/decreases
When the wing area increases/decreases, the gust load factor increases/decreases.
Increasing/decreasing the aspect ratio of the wing will increase/decrease the gust load factor.
Increasing/decreasing the speed will increase/decrease the gust load factor.
When the mass decreases/increase, the gust load factor increases/decrease.
When the altitude decreases/increase, the gust load factor increases/decrease.
When the wing loading decreases/increase, the gust load factor increases/decrease

Wing planforms **least** affected by **turbulence** are **Swept**, **low aspect** **ratio.**
Wing shape or wing **characteristic** is the **least** sensitive to **turbulence swept wings.**

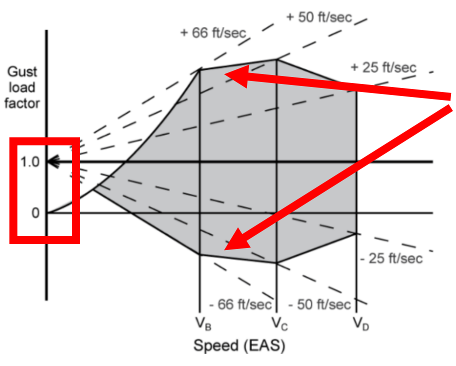
Flight in severe turbulence may lead to a stall and/or structural limitations being exceeded.

An aircraft flying at a given EAS is subject to a positive gust of 50 kts. An **increase** in positive **g-force** "felt" by the aircraft is **More with a high aspect ratio straight wing.**



The **extreme right limitation** for both V-n (gust and manoeuvre) diagrams is created by the speed **V_D**

In **gust and manoeuvre load diagrams**, the most limiting speed is **V_D**



The gust load diagram has a symmetrical shape with respect to the n = 1 line for speeds above V_B.

In a gust load diagram and its impact on the gust load factor **The steeper the slope of the lift curve, the greater the gust load factor.**

Gust values used in the V-n diagram are **50 ft/sec, 66 ft/sec, 25 ft/sec.**

A correct **design gust value** is **66 ft/sec at V_B.** **B** **high movement with gust at high speed**
A correct **design gust value** is **50 ft/sec at V_C.** **C**
A correct **design gust value** is **25 ft/sec at V_D.** **D** **less movement with gust at high speed.**

All gust lines in the **gust load diagram** originate from a point where the **speed = 0**, **load factor = +1.**

V_{RA} is the recommended turbulence penetRAtion air speed.

V_B is the design speed for **maximum gust intensity.** (Vblow)

The load factor in turbulence may fluctuate above and below 1 and can even become negative.

LOAD FACTOR = LIFT / WEIGHT **balance 1g = straight and level**

New V_S = oldV_S x $\sqrt{n}$

Load factor increase/decrease by half of V_S change

Load factor = C_L / oldC_L

A fundamental **difference** between the **manoeuvring limit load factor** and the **gust limit load factor** is, that **the gust limit load factor can be higher than the manoeuvring limit load factor.**

An aeroplane maintains straight and level flight **at a speed of 1.2V_S**. If a **vertical gust causes a load factor of 1.6**, the **load factor n** caused by the same gust **at a speed of 2V_S** would be **not relevant. The gust at 1.2 V_S would cause a stall. (New V_S = oldV_S x $\sqrt{n}$)**

An aeroplane maintains straight and level flight **at a speed of 1.5V_S**. If a **vertical gust causes a load factor of 2.5**, the **load factor n** caused by the same gust **at a speed of 1.2V_S** would be **irrelevant, since the aeroplane would already be in a stalled condition at 1.5V_S with that gust. (New V_S = oldV_S x $\sqrt{n}$)**

An aeroplane maintains straight and level flight **at a speed of 1.1V_S**. If a **vertical gust causes a load factor of 2.1**, the load factor n caused by the same gust **at a speed of 2V_S** would be **irrelevant, since the aeroplane would already be in a stalled condition at 1.1V_S with that gust. (New V_S = oldV_S x $\sqrt{n}$)**

An aeroplane maintains straight and level flight **at a speed of 2V_S**. If a **vertical gust causes a load factor of 2**, the **load factor n** caused by the same gust **at a speed of 1.3V_S** would be **1.65**
(speed - 0,7, gust load factor -0,35)

An aeroplane maintains straight and level flight **at a speed of 1.8V_S**. If a **vertical gust causes a load factor of 1.9**, the **load factor n** caused by the same gust **at a speed of 2V_S** would be **2.00**
(speed + 0,2, load factor +0,1)

An aeroplane maintains straight and level flight **at a speed of 1.7V_S**. If a **vertical gust causes a load factor of 2.7**, the **load factor n** caused by the same gust **at a speed of 2V_S** would be **3.00**
(speed + 0,3, load factor +0,15)

In straight and level flight at 300 kts assuming the angle of attack remains constant initially and the load factor reaches a value of approximately **1.2 the speed will have increased by 30 kts. (New V_S = V_S x $\sqrt{n}$)**

An aeroplane in straight and level flight at 200 kts is subjected to a sudden disturbance in speed. Assuming the angle of attack remains constant initially and the load factor reaches a value of 1.44 **the speed will have increased by 40 kts. (New V_S = oldV_S x √n)**

An aeroplane in straight and level flight at 100 kts is subjected to a sudden disturbance in speed. Assuming the angle of attack remains constant initially and the load factor reaches a value of 1.96 **The speed will have increased by 40 kts. (New V_S = oldV_S x √n)**

An aeroplane in straight and level flight at 100 kt is subjected to a sudden disturbance in speed. Assuming the angle of attack remains constant initially and the **load factor** reaches a value of **2.25** **The speed will have increased by 50 kt. (New V_S = oldV_S x √n)**

An aeroplane in straight and level flight at 100 kts is subjected to a sudden disturbance in speed. Assuming the angle of attack remains constant initially and the load factor reaches a value of 1.44 **The speed will have increased by 20 kts. (New V_S = oldV_S x √n)**

An aeroplane in straight and level flight at 100 kts is subjected to a sudden disturbance in speed. Assuming the angle of attack remains constant initially and the load factor reaches a value of 1.69 **the speed will have increased by 30 kts (New V_S = oldV_S x √n)**

An aircraft is in straight, level flight has a C_L of 0.42, and a 1° increase in angle of attack would **increase the C_L by 0.1**. Following a gust that increases the angle of attack by 3°, what **load factor** would the aircraft be **subject** to **1,7** **(load factor increase by 0.3, so 0.42+0.3 = lift 0.7 / weight 0,42 = 1.7)**

The lift coefficient (C_L) of an aeroplane in steady horizontal flight is 0.4. Increase of angle of attack of 1° will increase C_L by 0.09. A vertical up-gust instantly changes the angle of attack by 5°. The load factor will be **2.13** **(load factor increase by 0.09x5, so 0.4+0.45 = lift 0.85 / weight 0,4 = 2.13)**

The lift coefficient (C_L) of an aeroplane in steady horizontal flight is 0.35. Increase in angle of attack of 1° will increase C_L by 0.079. A vertical up-gust instantly changes the angle of attack by 2°. The load factor will be **1.45** **(load factor increase by 0.079 x 2 = 0.158 + 0.35 = 0.508/0,35 = 1.45)**

An aeroplane flying at 200 kts in straight and level flight is subjected to a disturbance that suddenly decreases the speed by 80 kts. Assuming the angle of attack remains constant, the load factor will initially **decrease to 0.36. (New V_S = oldV_S x √n) → √n = (New V_S / V_S)**

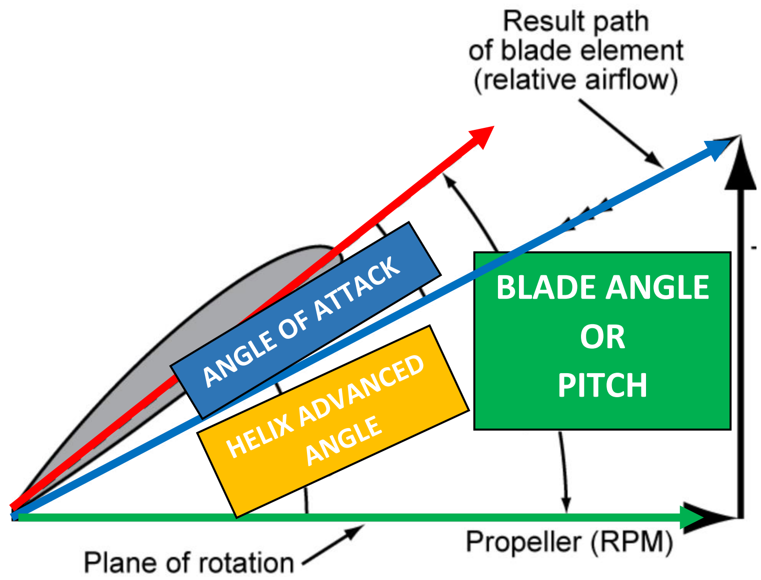
An aeroplane flying at 200 kts in straight and level flight is subjected to a disturbance that suddenly increases the speed by 80 kts. Assuming the angle of attack remains constant, the load factor will initially **increase to 1.96. (New V_S = oldV_S x √n) → √n = (New V_S / V_S)**

Extending flaps in turbulence reduces the stall speed but will reduce the margin to structural limitations.

The **gust load factor** due to a vertical up-gust **increases** when the **gradient of the C_L**-alpha graph **increases.**

Propellers

Conversion of Engine Torque to Thrust



The blade angle of a propeller is the angle between **the propeller chord and the plane of rotation of the propeller.**

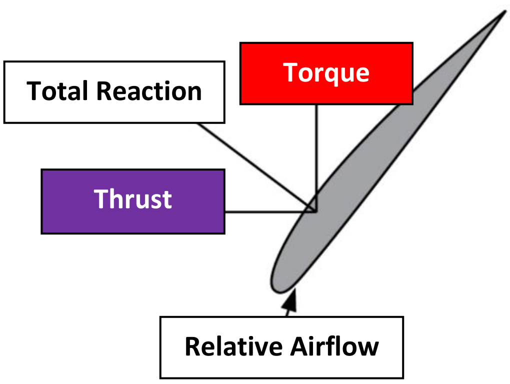
The angle of attack **of a propeller blade** element is the angle between the **blade element chord** line and the **resultant** air speed vector.

Propeller blade angle of attack is the angle between the chord and the relative airflow.

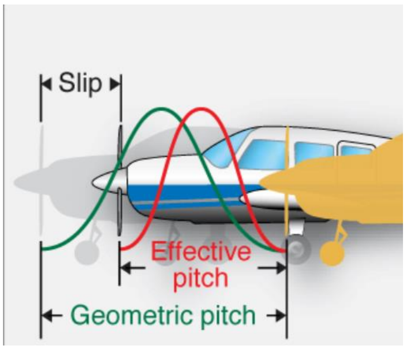
The **difference** between a propeller's blade angle and its angle of attack is called the helix angle.

The **blade angle** of a propeller is usually **referenced** at 75 % **of blade radius**.
The **reference** section of a propeller blade with radius R is usually taken at a distance from the propeller axis equal to 0.75 **R.**

Forces acting on a blade



The **forces** acting on a propeller are thrust **and** torque.



Geometric pitch is the theoretical distance a propeller blade element would travel in a forward direction during one revolution.

The geometric pitch of a propeller is the theoretical distance **a propeller would** advance **in one revolution at** zero **blade** angle of attack.

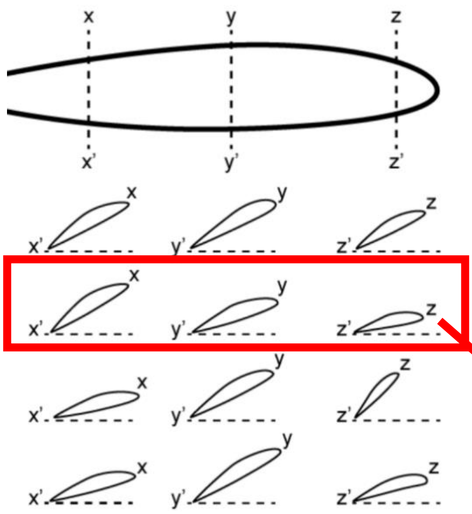
The **effective pitch** of a propeller is the **actual distance** **a propeller advances in one revolution.**
The **effective pitch** of a propeller depends on AoA. Change angle change distance and efficiency

Propeller slip is the difference between the **geometric pitch and the effective pitch.**

Constant-speed propellers deliver **better performance** than fixed-pitch propellers because they **operate at a relatively high propeller efficiency over a wider speed range than a fixed pitch propeller.**

If the **RPM lever** of a constant speed propeller **is moved forward** during a glide with idle power and whilst maintaining constant airspeed, the propeller **pitch** will **decrease and the rate of descent will increase. (forward in take off)**

Propeller efficiency may be defined as the ratio between **usable (power available) power of the propeller and shaft power.**



The correct sequence of cross-sections representing propeller blade twist is **sequence 2.**

From root to the extremities the blade angle of a fixed pitch decreases because the extremities run much faster than root

A propeller **blade is twisted** from root to tip **to maintain a constant angle of attack along the whole length of the propeller blade.**

The **twisting** of a propeller blade from root to tip has been made to **provide a constant angle of attack from root to tip**

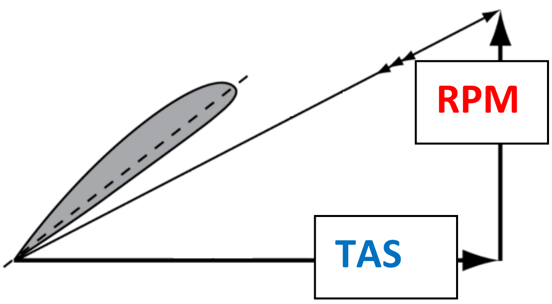
Propeller **blade twist** is the **varying of the blade angle from the root to the tip of a propeller blade.**

A constant speed propeller reduces fuel consumption over a range of cruise speeds.
A constant speed propeller improves take-off performance as compared with a coarse fixed pitch propeller.
A constant speed propeller improves propeller efficiency over a range of cruise speeds.
A constant speed propeller maintains near maximum efficiency over a wider range of aeroplane speeds than a fixed pitch propeller.
A propeller with a small blade angle is referred to as being in fine pitch. (fine=fino)
A propeller with a large blade angle is referred to as being in coarse pitch. (coarse=grosso)
A cruise propeller has a greater geometric pitch compared with a climb propeller.
A coarse pitch propeller is less efficient during take-off and in the climb, but more efficient in the cruise, when compared with a fine pitch propeller.
At a given RPM the propeller efficiency of a fixed pitch propeller is maximum at only one value of TAS

~~A propeller with little blade twist is referred to as being in fine pitch.~~
~~A propeller with significant blade twist is referred to as being in coarse pitch.~~

A **fixed pitch** propeller is usually **at too coarse an angle for take-off**. **(fixed is for cruise, coarse)**
During **Take-off run angle of attack** is **highest** in a **fixed pitch propeller's**
For a **fixed-pitch propeller** the **blade angle of attack** will **decreases during the take-off.**
For a **fixed-pitch propeller**, the **blade angle** of attack **can become negative during high-speed idle descent**. A **fixed** pitch **propeller's angle of attack** is **lowest High-speed glide.**

For a **fixed-pitch** propeller designed for cruise, the angle of attack of each blade, measured at the **reference** section **is optimum when the aircraft is in a stabilized cruising flight.**

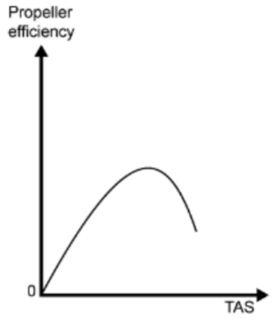


For a **fixed-pitch propeller** in flight at a given TAS, the **blade angle of attack** will **decrease if RPM decreases.**
For a **fixed-pitch propeller** in flight at a given TAS, the **blade angle of attack** will **increase if RPM increases.**
The **angle of attack** of a fixed pitch propeller blade **increases** when **forward velocity decreases and RPM increases.**

With a **fixed pitch** propeller **increasing speed** will **decrease** propeller angle of attack. **Increasing** power and therefore propeller **RPM** will **increase** propeller angle of attack
Assuming that the **RPM remains** constant throughout, the **angle of attack** of a fixed pitch propeller will **increase with decreasing airspeed.**
For a given RPM of a fixed pitch propeller, the **blade angle of attack** will **decrease when the TAS increases.**
For a given RPM of a fixed pitch propeller, the **blade angle of attack** will **increase when the TAS decreases.**
Assuming that the **RPM remains constant** throughout, the **angle of attack** of a fixed pitch propeller will **decrease with increasing airspeed.**

A typical **fixed pitch** propeller (e.g. C-172) is designed to achieve its optimum angle of attack at **cruise speed.**

Running an engine fitted with a fixed pitch propeller at **full throttle** with the **aircraft stationary** and **nose into strong wind** will result in **higher RPM than in still air.**

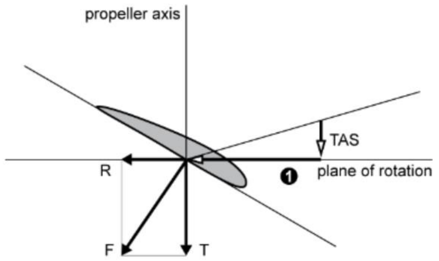


A typical **curve representing propeller efficiency** of a **fixed pitch** propeller versus TAS at **constant RPM**

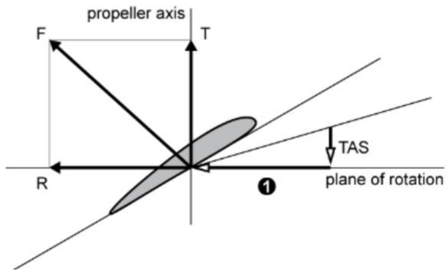
The variation of **propeller efficiency** of a **fixed pitch** propeller with TAS at a given RPM is shown in figure

A rotating propeller blade produces force F that may be resolved into two components:

- a **force T perpendicular** to the **plane of rotation** (**thrust**)
- a **force R generating a torque** absorbed by engine power



The diagram representing a rotating propeller blade during **reverse** operation



The diagram representing a rotating propeller blade **during the cruise**

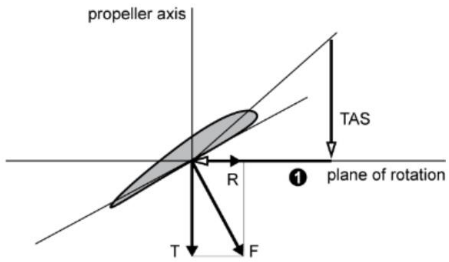
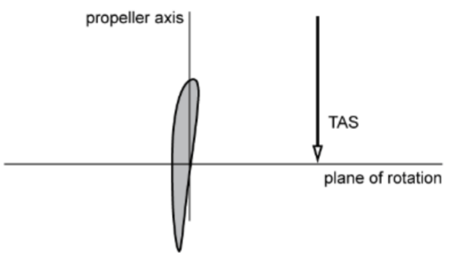


Diagram 4

The diagram representing a **windmilling propeller** (propeller is driven by the wind and not by the engine)



The diagram representing a **feathered propeller**

With a **constant speed propeller**. If the aeroplane **speed decreases** while **manifold pressure** remains **constant**, propeller **pitch** will **decrease** and propeller **torque** will **remain constant**.

With a **constant speed propeller**. If the aeroplane **speed increases** while **manifold pressure** remains **constant**, propeller **pitch** will **increase** and propeller **torque** will **remain constant**.

If **ice builds up on a propeller** during flight **The generated thrust will decrease.**
Propeller icing increases blade element drag and reduces blade element lift.
Propeller icing reduce propeller efficiency
A reduction in efficiency up to 20 % can be encountered if ice is present on the propeller blades.

The **first action** in event of **propeller runaway (overspeed condition)**, should be to **close the throttle.**

During a **glide with idle power** and **constant IAS**, if the **RPM lever** of a constant speed propeller is pulled **back** from its normal cruise position, the propeller **pitch** will **increase** **and the rate of descent will decrease.**

Propeller **efficiency** is **the ratio of power available (Thrust × TAS) to shaft power (Torque × RPM).** Propeller **efficiency** can be expressed as **power output divided by power input.**

About a **constant speed propeller** the **blade angle increases** with **increasing** aeroplane **speed.**

The **pitch angle** of a constant-speed propeller varies **slightly** in medium horizontal **turbulence**

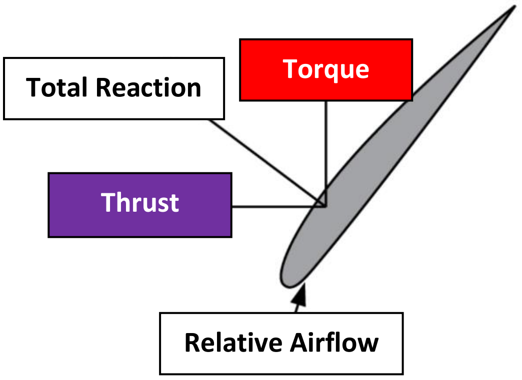
The **aerodynamic loads** on a propeller which produce **forward thrust** will tend to **bend the tips forward.**

Propeller torque is caused by **the forces caused by the airflow on the propeller.**

If the **propeller pitch**, within its constant speed range and at constant power, **during descent** at **constant IAS is decreased**, the aeroplane **lift to drag ratio** will **decrease and the rate of descent will increase.**
If the **propeller pitch**, within its constant speed range and at constant power, **during descent** at **constant IAS is increased**, the aeroplane **lift to drag ratio** will **increase and the rate of descent will decrease.**

For an aeroplane equipped with a **two-position variable pitch** propeller it is advisable to select a **fine pitch for take-off and climb.**

A **variable pitch propeller** during take-off will move towards **Fine pitch to ensure that the engine can develop its maximum power.**



For any propeller **thrust is the component of the total aerodynamic force on the propeller parallel to the rotational axis.**

Thrust parallel to rotational axis, airflow
Torque perpendicular to rotational axis,

Engine Failure

A feathered propeller causes less drag than a windmilling propeller.

A **windmilling** propeller **produces drag instead of thrust.**
The **greatest drag** produced by the variable pitch propeller on a piston engine will occur when the propeller is **windmilling.**
An engine failure can result in a **windmilling** propeller (with **more drag**) and a **non-rotating propeller** (with **less drag**). **The windmilling drag is much higher than for a feathered propeller.**

If the **propeller pitch** of a **windmilling** propeller **is decreased during a glide** at constant IAS the **propeller drag** in the direction of flight will **increase and the rate of descent will increase.**

If the **propeller pitch** of a **windmilling** propeller **is increased during a glide** at constant IAS the **propeller drag** in the direction of flight will **decrease and the rate of descent will decrease.**

When compared with a non-feathered propeller, a feathered propeller improves the handling of a multi-engine aeroplane with one engine inoperative.

When the blades of a propeller are in the feathered position the drag of the propeller is then minimal.
The blade angle of a feathered propeller is approximately 90 degrees.
With a propeller **feathered there will be minimum drag on the propeller.**

Design Features for Power Absorption

Increasing tip speed to supersonic speed increases propeller noise.
Increasing tip speed to supersonic speed decreases propeller efficiency.

Power absorption increases if the mean chord of the blades increases.
Assuming blade shape does not change, power absorption increases if the number of blades increases.
Increasing the **number of propeller blades** will **increase the maximum absorption of power.**

The **more blades a propeller has**, the **more power** it is able to **absorb**. The limitation on blade number from an aerodynamic standpoint is **the loss of efficiency of one blade if it follows to the path of the preceding blade too closely.**

Disadvantages of **excessive** increasing of the **number** of propeller **blades is Decreased propeller efficiency.**

The **propeller noise** can be **minimised** by **increasing the number of blades.**

Propeller noise increases when the blade tip speed increases.
For a given engine and propeller blade shape, an increase in the number of propeller blades allows for a reduction in propeller noise.

Increasing the **camber** on propeller **blades** will, if all else is the same **increase the power absorption capability.**

If the **chord** of its propeller **blades increases Propeller efficiency decreases and power absorption increases. (larger blades are more powerfull but less efficient)**

Propeller "solidity" (area ratio) is the ratio of **the total frontal area of all the blades to S.**
To calculate the "**area ratio**" (solidity) of the **propellers** you divide **Area of all propeller blades to the total circular surface.**
If the **number of blades** on a propeller is **increased The solidity of the propeller will increase.**

Secondary Effects of Propellers

Counter rotating propellers have the effect of **cancelling out the torque and gyroscopic effects.**
Counter rotating propellers is the name of the propeller system, where the **propellers** of a twin-engine aeroplane are **rotating in opposite directions**

CONTRA Rotating Propeller is the name of the propeller system having **two propellers rotating** in the **opposite direction** on **the same shaft**

Torque effect produce roll
Slipstream effect produce yaw.
Torque+ Slipstream effects are part of the Gyroscopic effect!
Regarding the **gyroscopic effect** of a **clockwise** rotating propeller 卐
Pitch up produces right yaw.
Right yaw produces pitch down.
Pitch down produces left yaw.
Left yaw produces pitch up.

A propeller **rotating clockwise** as seen **from the rear** tends to **rotate the aircraft** to the **left around the vertical axis, and to the left around the longitudinal axis.**

A propeller is **turning to the right** when viewed from behind. The **asymmetric blade effect** in the **climb** at **low speed** will **yaw the aeroplane to the left.**
The **asymmetric blade effect** on a single engine aeroplane with a **clockwise** rotating propeller **produces left yaw.**

Propeller gyroscopic effect occurs during aeroplane yaw changes.
Propeller gyroscopic effect occurs during aeroplane pitch changes.
Propeller gyroscopic effect is most noticeable during low speed flight at high propeller RPM.
Constant attitude not give gyroscopic effect.

Gyroscopic precession of the propeller is induced by **pitching and yawing.**

Asymmetric blade effect increases when engine power is increased
Asymmetric blade effect increases when the angle between the propeller axis and the airflow through the propeller disc increases.

Asymmetric propeller blade **effect** is mainly induced by **the inclination of the propeller axis to the relative airflow.**

The **torque reaction** of a rotating fixed pitch propeller will be greatest at **low aeroplane speed and maximum** engine **power.**

Propeller RPM would change the **magnitude** of the **gyroscopic precession** effect of the propeller

Given an aeroplane with a propeller turning **anti-clockwise** as seen from behind, the **torque effect** will tend to **roll the aeroplane to the right.**
Given an aeroplane with a propeller turning **clockwise** as seen from behind, the **torque effect** during the take-off run will tend to **roll the aeroplane to the left.**

In **twin engine** aeroplanes with propellers turning **clockwise the left engine is the critical engine.**

During the **take-off roll**, when the pilot **raises the tail in a tail wheeled** propeller driven aeroplane, the **additional** aeroplane **yawing tendency** is due to the effect of **gyroscopic precession.**

For a **tail wheel aircraft** with a **propeller turning to the right** (when viewed from behind), at the start of the take-off run, **asymmetric blade effect** causes **Yaw to left.**

Given **two** identical aeroplanes with **wing mounted engines**, **one** fitted with **jet engines** and the **other with counter rotating propellers. In case of engine failure. The propeller aeroplane has more roll tendency. (jet engine have less gyroscopic effect)**

In a single engine aircraft with a **clockwise rotating propeller** (seen from behind), a **left yaw** is generated due to **the slipstream, striking the fin on the left side.**

For a single engine aircraft with a propeller turning **clockwise** as seen from behind, the **slipstream** rotation will cause **yaw to the left.**

To **counteract** the effect of **slipstream** on a single engined aircraft the **fin should be placed as far as possible from the propeller. (tail, vertical fin as far as possible)**

Flight Mechanics
Forces Acting on an Aeroplane

In steady **level flight** the **load factor** is **1.0**

Dividing lift by **weight** gives **load factor.**

The four **forces acting** on an aeroplane in level flight are **thrust, lift, drag and weight.**
The four forces of **lift**, **weight**, **thrust** and **drag** in level flight act through **the CG.**
Weight acts **parallel to the gravitational force.**

The **forces of lift and drag** on an aerofoil are, respectively, **normal and parallel** to the **relative airflow.**

Regarding the **opposing forces** acting on an aeroplane in steady state **level flight The opposing forces are equal.**

An aircraft is in level flight at **low speed** and **high angle of attack**, the normal axis is **nearly vertical.**

During the **glide**, the **forces acting** on an aircraft are **lift, drag and weight.**

In a **straight steady descent Lift is less than weight, load factor is less than 1.**

The **descent angle** of a given aeroplane in a **steady wings level glide** has a fixed value for a certain combination of **configuration and angle of attack.**

For a **straight** and **steady** descent **The steeper the descent angle, the greater the difference between total weight and lift.**

When **gliding**, the speed which will give the **minimum rate of descent** is **less than the speed for maximum glide range.**

Regarding the **forces acting** on an aircraft in a steady-state descent **The sum of all forward forces is equal to the sum of all rearward forces.**

During a straight steady descent, **lift** is **less than weight, because lift only needs to balance the weight component perpendicular to the flight path.**

The **maximum glide range** of an aircraft will depend on wind and **the ratio to lift to drag which varies according to angle of attack.**

True airspeed for **minimum sink rate** is (i) **less than** **true airspeed** for (ii) **minimum glide angle**, at a given altitude

A tailwind with constant aeroplane mass compared with zero wind increases the maximum **ground distance during a glide** with zero thrust
A tailwind increases the maximum **ground distance during a glide**
A headwind with constant aeroplane mass compared with zero wind decreases the maximum **ground distance** during a glide with zero thrust

A decrease in mass increases the maximum **duration of a glide**

To cover the **greatest distance** when gliding, **the gliding speed** must be **the one that gives the lowest total drag.**
Maximum gliding distance of an aircraft is obtained when **induced drag and parasite drag are equal.**

The **lift to drag ratio** determines the **horizontal glide distance from a given altitude at zero wind and zero thrust.**

Factors that determine the **distance travelled** over the ground of an aeroplane **in a glide are The wind and the lift/drag ratio.**

An aircraft will achieve the **greatest** possible **gliding distance** when **it is flown at a speed corresponding to the highest lift/drag ratio.**
Assuming zero wind, an aeroplane in a straight **steady glide** will achieve the **minimum glide angle** at **(C_L/C_D) MAX**

If an airplane would perform a **vertical↑ climb** with **constant speed Thrust = Weight + Drag**
During a steady climb (flight path angle **90°**)↑ **thrust equals weight plus drag.**

In a **climb** the **weight component** along the **flight path is balanced** by **thrust**

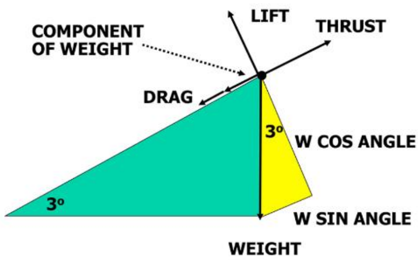
An **aeroplane climbs** to cruising level with a constant pitch attitude and **maximum climb thrust** (assume no supercharger). (gamma = flight path angle) **gamma decreases, angle of attack increases, IAS decreases.**

In a straight **steady climb** the **thrust must** be **greater than the drag because it must also balance a component of weight.**

During a straight **steady climb** and with the **thrust** force **parallel** to the **flight path, lift is the same as during a descent at the same angle and mass.**

During a straight steady **climb**:
lift is less than weight.
load factor is less than 1.

Cos is longer edge
Sin is smaller edge



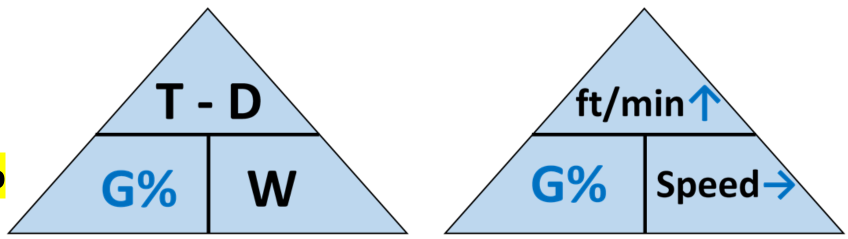
Ignoring thrust effects in a steady straight climb at **a climb angle "gamma"**, the **lift** of an aeroplane with weight W is **W × cos (gamma)**

the **thrust** of an aeroplane with weight W is **W × sin (gamma)**

In a straight **steady climb thrust equals the sum of drag and the weight component along the flight path, and lift equals the weight component perpendicular to the flight path.**

During a **climbing turn** to the right the **angle of attack of the left wing is larger than the angle of attack of the right wing. (higher wing = higher lift = higher angle)**

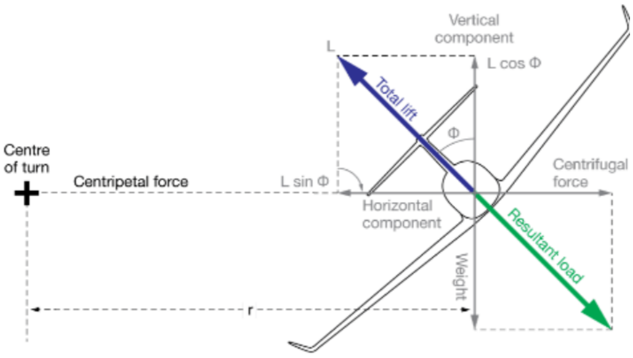
Drag = W ÷ ratio



Aeroplane **mass**: 50 000 kg
Lift/Drag ratio: 12
Thrust per engine: 21 000 N
Assumed **g**: 10 m/s^2
For a straight, steady, wings level climb of a **four-engine** aeroplane, the **one-engine inoperative climb gradient** will be **4.3%** **Drag = W ÷ ratio**

Aeroplane **mass**: 50 000 kg
Lift/Drag ratio: 12
Thrust per engine: 28 000 N
Assumed g: 10 m/s2

For a straight, steady, wings level climb of a **three-engine** aeroplane, the **one-engine-inoperative climb gradient** will be **2.9%**   **Drag = W ÷ ratio**



Steep banked turn

A centripetal force is the horizontal component of the lift force that equals the weight and makes the airplane to follow a curved path

In a **steady banked turn** the **lift** will **equal the resultant of weight** and **centripetal force.**

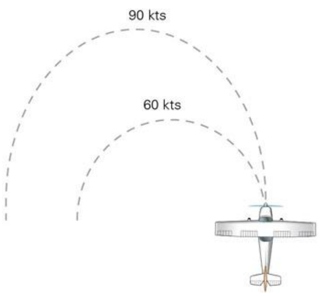
Turning motion in a steady, level co-ordinated turn is created by **the centripetal force.**

During a correctly **balanced turn the lift force provides a centripetal force and a force that opposes the weight of the aircraft.**

The **necessary** normal **acceleration** to make an **aircraft turn is the horizontal component of lift.**

To **maintain altitude and airspeed** when performing a **co-ordinated turn** in a jet aircraft pilot **Increase angle of attack and thrust.**

An aeroplane is performing a co-ordinated level turn with a constant 30° bank angle. If the **airspeed** is **increased** the **rate of turn** will **decrease** and the **radius** of turn will **increase**



In order to fly **a rate one turn** at a **lower airspeed**, the **bank angle** must be **decreased** and the **turn radius will decrease.**

In order to fly **a rate one turn** at a **higher airspeed**, the **bank angle** must be **increased** and the **turn radius will increase.**

Comparing two identical **aeroplanes**, but **one is heavier** than the other. The **minimum** possible **radius of turn** will be **smaller for the lighter aeroplane.**

An aeroplane enters a **horizontal turn with a load factor n=2** from straight and level flight whilst **maintaining constant indicated airspeed**. The **lift doubles.**

In a steady co-ordinated **horizontal turn**, **lift** is **greater than in straight and level flight, because it must balance the weight and generate the centripetal force.**

In order to **maintain constant speed** during a level, **co-ordinated turn,** compared with straight and level flight, the pilot must **increase thrust/power and angle of attack.**

Compared with level flight, the **angle of attack** must be **increased in** a steady, co-ordinated, **horizontal turn to compensate for the reduction in the vertical component of lift.**

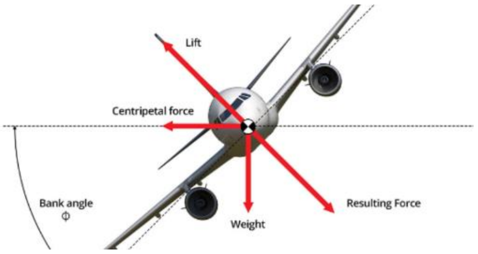
The **bank angle** in a rate-one **turn** depends on **TAS.** **(bankΦ = TAS+ 7)**

A **turn is co-ordinated When the longitudinal axis of the aeroplane at the CG is tangential to the flight path.**

If a pilot initiates a **coordinated turn** from a straight and level flight and if maintains current altitude and thrust **The speed will decrease.**

In a steady, **horizontal, co-ordinated turn thrust equals drag**, because there is **equilibrium of forces** along the **direction of flight.**

In a steady, **level, co-ordinated turn** the **rate of turn** depends upon the **TAS** and **angle of bank.**



In a **co-ordinated horizontal turn**, the magnitude of the **centripetal force** at 45° of bank **is equal to the weight of the aeroplane.**

Respect to rate and radius of turn for an aeroplane flown in a co-ordinated turn at a constant altitude **For a specific angle of bank and airspeed the rate and radius of turn will not vary.**

For **shallow** flight path angles in straight and steady flight, the following formula can be used **$\sin(\gamma) = T/W - C_D/C_L$ (IMPARARE A MEMORIA, RISPOSTE UGUALI)**

For an aircraft in level flight, if the wing **centre of pressure is aft** of the centre of gravity and there is no thrust-drag couple, the **tailplane** load must be **directed downwards.**

In a **skidding turn** (the nose pointing inwards), compared with a coordinated turn, the bank angle is **too small** and the "ball" or "slip indicator" is displaced **towards the high wing.**

In a **slipping turn** (nose pointing outwards), compared with a co-ordinated turn, the bank angle is **too large** and the "ball" or "slip indicator" is displaced **towards the low wing**

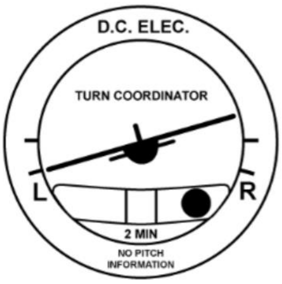
An aeroplane's **flight path angle** is defined as the angle between its **speed vector and the horizontal plane.**

Slip/Ball is where your body is pushed by G force

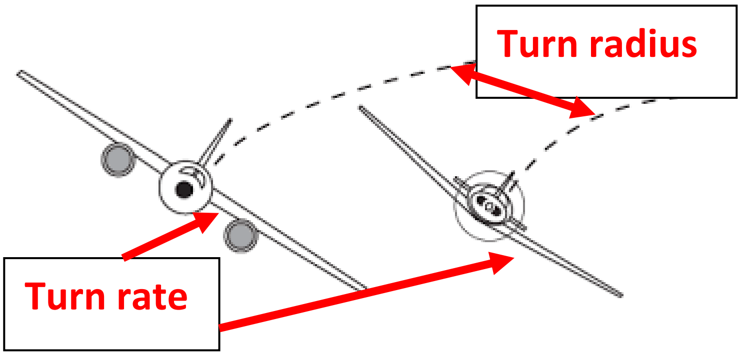
An aeroplane performs a turn. If the **turn and slip indicator needle** is to the **left** and the **ball** is displaced to the **right**, the correct control input to co-ordinate the turn **Less left rudder.**

An aeroplane performs a turn. **The turn indicator shows a right** turn while the **slip indicator is left** of neutral. One way to co-ordinate the turn is to **apply more right bank.**

An aeroplane performs a **right turn**, the **slip indicator is left of neutral**. One way to co-ordinate the turn is to apply **more left rudder.**



A **left turn** with **not enough bank**



The same aeroplane with the same bank angle and speed**, but at a lower mass** will **turn with the same rate of turn**

An aeroplane performs a steady horizontal, **co-ordinated turn** with 45 degrees of bank at 230 kts TAS.
The same aeroplane with the same bank angle and speed, **but at a higher mass will turn with the same radius, but might stall.**

$$\text{Turn radius m} = (TAS \times 0.5144)^2 \div (g \times \tan\Phi)$$

At a **true airspeed** of 300 kts and in a 45° **bank** level turn, the **radius** of turn would be (assume a value of 10 m/sec2 for g) **2 381 m**

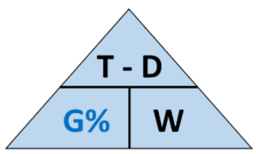
The approximate **radius of** a steady horizontal **co-ordinated turn** at **a bank angle of 45°** and a **TAS** of 200 kts is **1 km**

The approximate **diameter** of a steady, level, **co-ordinated turn** with a bank **angle of 30°** and a speed (**TAS**) of 500 kts **23 km** **(diameter is 2 x turn radius)**

During a straight **steady climb** with a **20% climb gradient**, the load factor is equal to **0.98 (load factor in climb and descend is just steady below 1)**

Descend angle $\sin\Phi$ = (Drag - Thrust) ÷ Weight
Climb angle $\sin\Phi$ = (Thrust - Drag) ÷ Weight
Lift = W ÷ cos Φ
Load Factor = 1 ÷ cosΦ

The thrust vector in a **descent** with a 30° angle, a **weight** of 11 350 N, a **lift** of 10 320 N and a **drag** of 11 010 N is **5335**



An aeroplane with a **mass** of 2 000 kg, is performing a **co-ordinated level turn** at a constant **TAS** of 160 kts and the **bank angle** is **60°.** The **lift** is approximately **40000 (60° means 2g, double of mass in N)**

An aeroplane with a **mass** of 4 000 kg is performing a **co-ordinated level turn** at a constant **TAS** of 160 kts and a **bank angle of 45°**. The **lift** is approximately **56 000 N**

Two identical aeroplanes A and B, **with the same mass,** are flying steady level co-ordinated 20° bank turns. If the TAS of A is 130 kts and the TAS of B is 200 kts (**Tas is higher**) **the turn radius of A is less than that of B.**

Heading change after **10 seconds** of an aeroplane performing a steady and coordinated horizontal rate one turn **30°** (A rate one turn or **standard turn** is a turn **360°** in **2 minute**)

Lift increase in a steady level turn at **45° angle of bank**, compared to straight and level flight by **41%** **(load factor = 1 ÷ cos Φ)**

With **a L/D ratio of 9**:1 and flying at 12 000 ft the glide range in still air would be approximately **18NM** **(1NM = 6080ft so** 12000/6080=1.973NM. 1.973 x 9 units = 17.76 NM. **)**

Asymmetric Thrust

V_{MC} for an aeroplane with unsupercharged engines **Decreases with increasing altitude. (less density, more control)**

V_{MCG} is the speed at which **directional control can be maintained at engine failure on take-off using the primary flying controls.**

During V_{MCG} determination, the nose wheel steering is not used.
In order to simulate a wet runway, nose wheel steering may not be used during V_{MCG} determination.
V_{MCG} must be determined using rudder control alone.
V_{MCG} is determined with the gear down.
During V_{MCG} determination, the CG should be on the aft limit.
During V_{MCG} determination, the lateral deviation from the runway centreline may be not more than 30 ft.
During V_{MCG} determination the aeroplane may not deviate from the straight-line path by more than 30 ft.

V_{MCG} is determined with the **nosewheel steering disconnected Because the value of V_{MCG} must also be applicable on wet and/or slippery runways**

V_{MCG} with **increasing** field **elevation** and **temperature Decreases, because the engine thrust decreases.**

V_{MCL} is the minimum control speed in the landing configuration.
V_{MCL} can be limited by the available maximum roll rate.

V_{MCL} is the minimum control speed in approach and landing.
V_{MCL} is **defined** with the aeroplane trimmed **For approach with all engines operating.**

V_{MCL} can be **limited** by:
roll rate.

When determining **V_{MCA} Maximum take-off thrust and 5° bank angle.**
V_{MCA} is certified with a **bank angle** of **not more than 5°** towards the operating engine (live engine low) because **although more bank reduces V_{MCA}, too much bank may lead to fin stall.**

V_{MCA} depends on the airport density altitude, and the location of the engine on the aeroplane (aft fuselage or wing).

V_{MCA} is the **minimum speed** at which **directional control** can be maintained when:
maximum take-off thrust was set and is maintained on the remaining engines.
a sudden engine failure occurs on the most critical engine.

As the bank angle is decreased from 5° to 0°, the value of V_{MCA} increases. (bank vs good engine)
When the bank angle is increased beyond 5°, there is an increasing risk of fin stall.
~~At any bank angle above 5 degrees, VMCA will decrease correspondingly.~~

For a given aeroplane **Airport elevation and temperature determine** the value of **V_{MCG}**

In order to **maintain straight and level flight** when you have just **lost your critical engine** you must **rudder and bank towards live engine.**

In general, **directional controllability** with one **engine inoperative** on a multiengine aeroplane is **adversely affected** by:
low temperature.
aft CG location.
low altitude. **Dense air is bad for gyroscopic effect**

In general, **directional controllability** with one **engine inoperative** on a multiengine aeroplane is **favorably affected** by
high temperature.
forward CG location.
high altitude.

If a twin engine **jet aeroplane** commences a take-off on runway 27 and the current wind is 210°/15 kts, **The left engine is the critical engine** in case of an engine failure during take-off run

If an aeroplane, that has experienced a **right engine failure** continues afterwards **in straight and level cruise** flight with wings level **Turn indicator is neutral, slip indicator is neutral.**

If an aeroplane, that has experienced a **left engine failure** and continues afterwards **in straight and level cruise** flight with wings level **Turn indicator is neutral, slip indicator is neutral.**
(Turn and slip indicator indicate a turn. **No turn means both neutral**. hsi indicate a bank angle.)

If an **engine fails** on a twin engined aeroplane a **yawing moment** will appear. When the **CG is located at the aft CG limit** will provide the **least** ability to **oppose this yawing** moment.

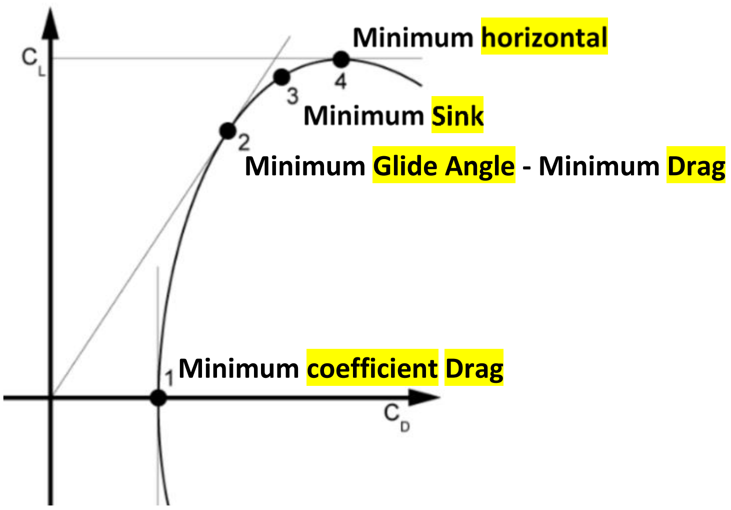
During a **take-off roll** with **a strong crosswind** from the right, a **four-engine** jet aeroplane with wing-mounted engines experiences an engine failure. The failure of **the right outboard engine** will cause the greatest control problem

During a **take-off roll** with **a strong crosswind** from the left, a **four-engine** jet aeroplane with wing-mounted engines experiences an engine failure. The failure of **the left outboard engine** will cause the greatest control problem

Equilibrium of moments about the normal axis is provided by rudder deflection.
Equilibrium of forces along the lateral axis requires either bank angle or sideslip or a combination of both. (about = turning around that axis. Along = maintain stable that axis)

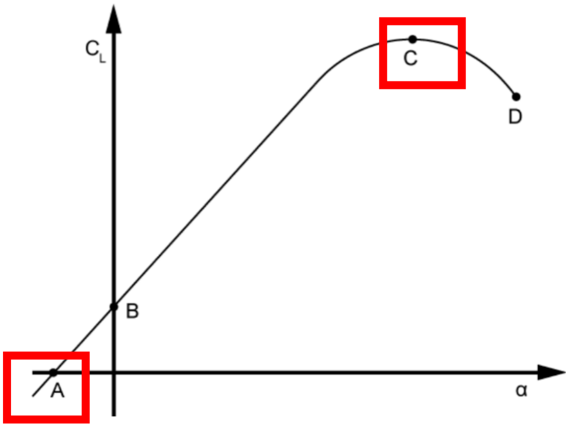
Particular Points on a Polar Curve

Given the following characteristic points on a jet engine aeroplane's polar curve:
Long range cruise (zero wind).
Maximum range cruise (zero wind).
Maximum lift to drag ratio.
Minimum rate of descent (assume zero thrust).
C_{LMAX}



At the **speed for minimum drag** (subsonic) **The gliding angle is minimum (assume zero thrust).**

The **speed for minimum glide angle** occurs at **(C_L/C_D)MAX**



The point showing **zero lift** is **A**

The point of **minimum horizontal flight** speed **C**