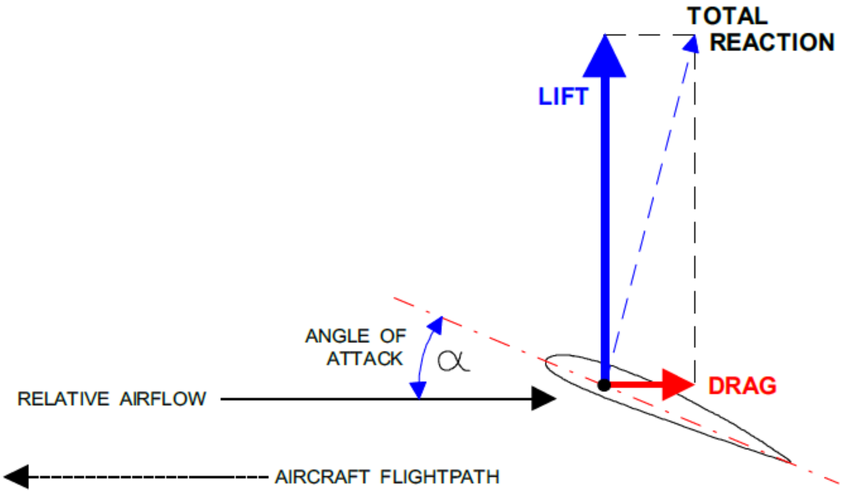
V. LIFT & DRAG RELATIONSHIP

In this part we will see the different methods to provide lift and drag with different aerofoils.

When an aircraft builds up a forward speed to generate Lift, it will also generate Drag, let's compare the Lift and Drag formulas:

$$L = ½\rho V^2.\ S.\ C_L$$

$$D = ½\rho V^2.\ S.\ C_D$$



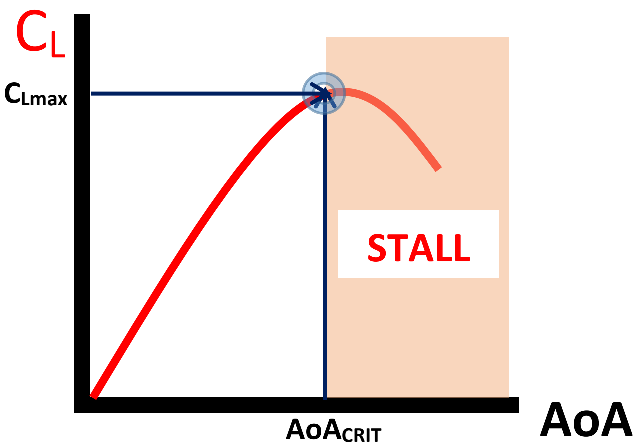
It can been seen that, no matter the factor is increased to generate more Lift, more Drag is also generated.

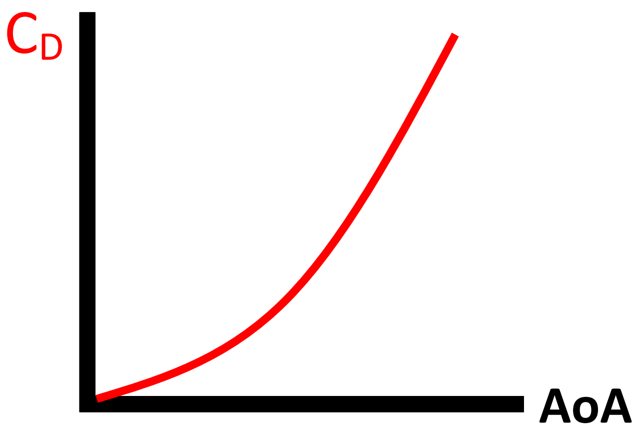
The Lift is necessary for an aircraft to fly, however the Drag is a penalty that will make the aircraft to decelerate and not fly fast, even if the engines could provide enough Thrust to act against, more fuel is required to be consumed.

In this part we will see how more Lift could be generated while maintaining the least Drag

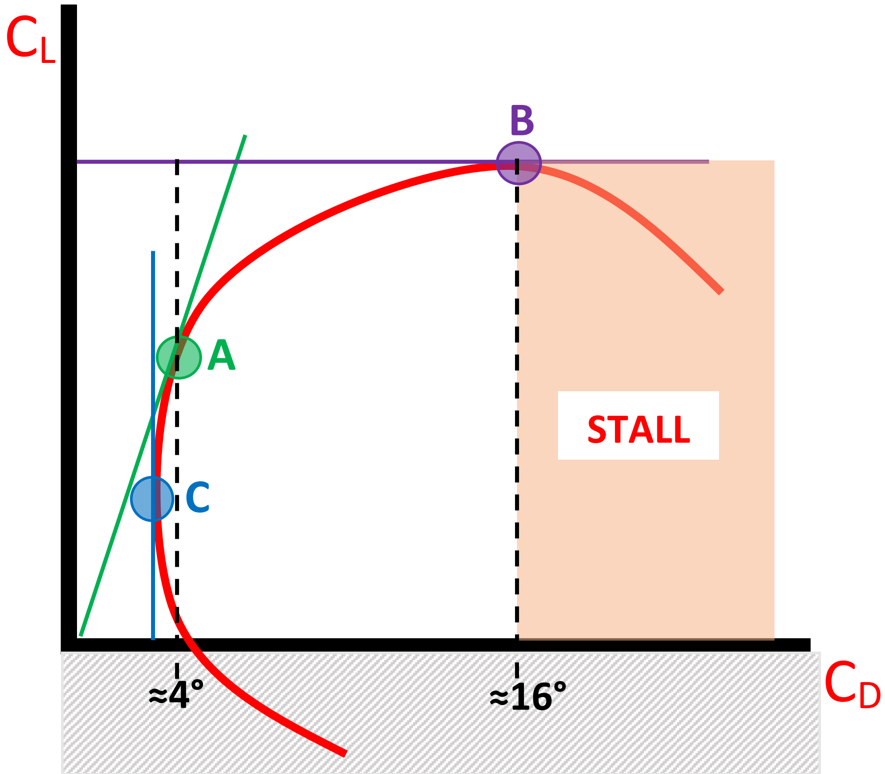
1) C_L vs C_D

For a given aerofoil, we compare its C_L and C_D for different AoA





Now if we project on a new curve, for given C_D the C_L for the same AoA, be obtain a curve called a **POLAR CURVE**



On this Polar Curve we can observe different points

A) When drawing the tangent to the Polar Curve, we obtain the AoA which gives the best relation between C_L and C_D, approximately at 4° AoA, it's the best C_L/C_D, which also means the best L/D

B) This is the highest point of the C_L, meaning this is C_{Lmax}, so this is the maximum AoA minimum airspeed that the aircraft is able to maintain, and it's usually neat 16° AoA

C) At this point, it's the minimum CD, lower Drag couldn't be made, so this is the minimum AoA or maximum speed that the aircraft could achieve.

The portion of the curve between the point **C** and **B** shows the aircraft's speed range that could be flown.

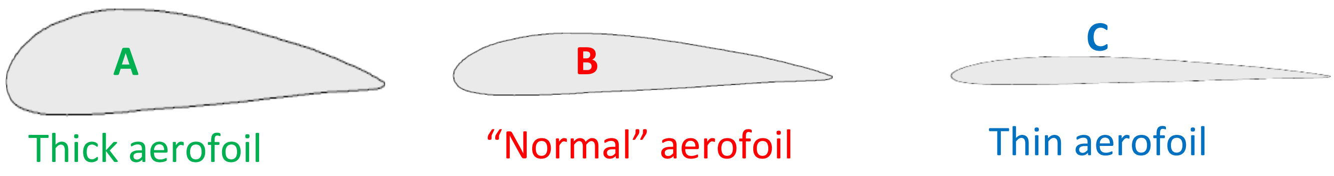
Note: The Polar Curve could be represented with different shapes depending on the aerofoil studied, but the principles are the same.

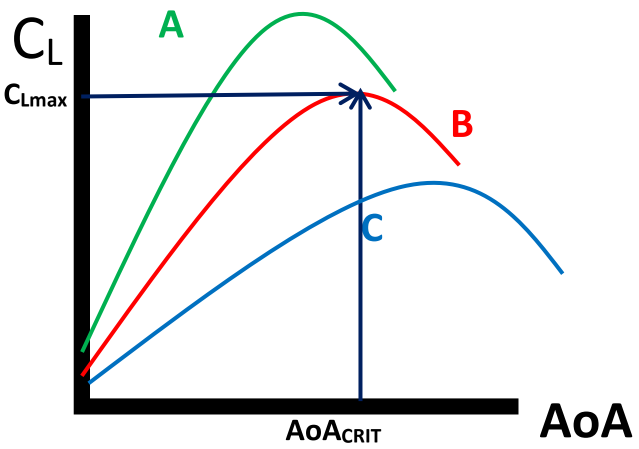
Different technics could be made to minimise the Drag, however the LIFT will also be affected.

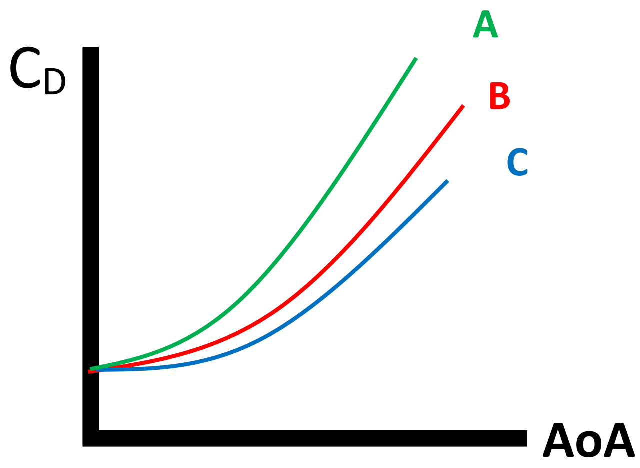
2) Aerofoil thickness and radius

By reducing the thickness or the radius of an aerofoil, there will be less acceleration of the airflow and so less static pressure reduced, resulting in less adverse pressure gradient and so less Drag, however there will be also less differential pressure and so less Lift

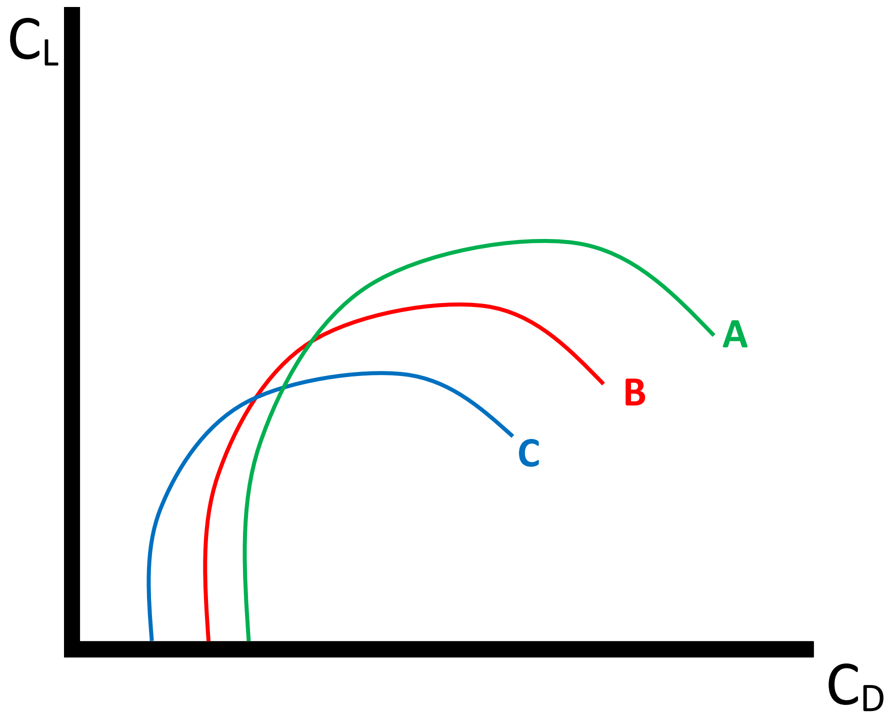
Remember! MORE LIFT = MORE DRAG, LESS LIFT = LESS DRAG







POLAR CURVE



Each airplane will have the aerofoil required for its own purpose of operation. The thick aerofoil will be chosen for airplanes that doesn't fly fast, because of their low speed they need a high lift aerofoil so they take off, fly and land a low speed, however they won't be able to fly fast because of the high drag generated. While the thin aerofoil will be chosen for high speed flights, but those aircraft will have poor lift capability at low speed forcing them to take-off and land at a high speed. Sometimes a compromise is made between the thin and large aerofoil is made to obtain an aerofoil that allow to fly slightly faster and generate lift at lower speed. The term normal is written "normal" because there is no normal aerofoil, the aerofoil is said to be thick or thin when compared to another one.

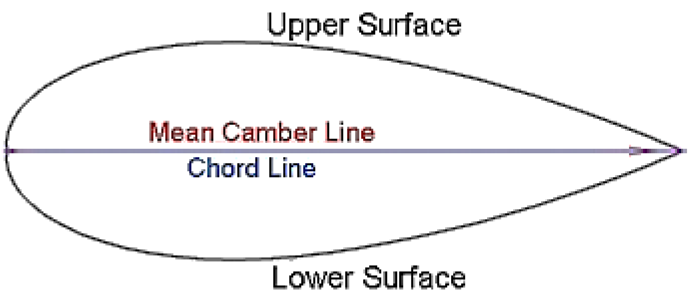
3) Aerofoil camber

The Coefficient of Lift C_L will depend on the camber of the aerofoil.

Symmetrical aerofoil or Zero camber aerofoil

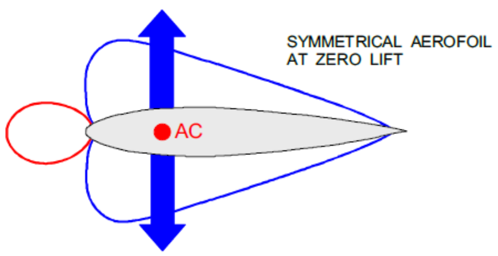
If the aerofoil is same on the lower surface and the upper surface, it's said to be a **symmetrical** aerofoil, the airflow accelerates at the same rate on the upper surface and on the lower surface of the aerofoil at an AoA=0°, so it will have no lift capability ($C_L = 0$) because the static pressure around the aeorfoil at AoA=0° is the same, the AoA must be increased (until AoA_{CRIT}) to have a lift capability ($C_L > 0$)

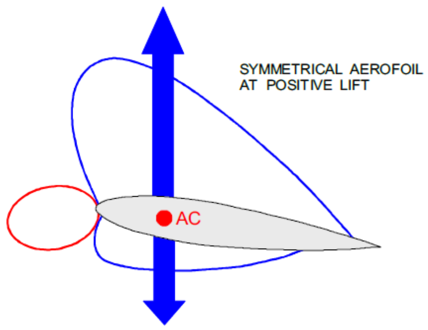
In this case, when drawing the chordline and the mean camberline of the aerfoil, the mean camberline will be the same as the chordline. The symmetrical aerofoil is said to have **zero camber.**



Since a zero camber aerofoil has no lift capability at AoA=0°; the AoA must be increased above 0° for the aerofoil to have $C_L > 0$. On the C_L curve, the curve will start at the origin of the axis (see graphs below)

Important note: When at zero angle of attack, the upper and lower surface forces are equal and located at the same point. With an increase in angle of attack, the upper surface force increases while the lower surface force decreases. A change in the magnitude of lift has taken place with no change in the CP position - one of the big advantages of symmetrical aerofoils. Thus, the rotation (pitching) moment about the AC for a symmetrical aerofoil will be zero at "normal" angles of attack – **The CP is always located at 25% of the chordline (or at the AC)**

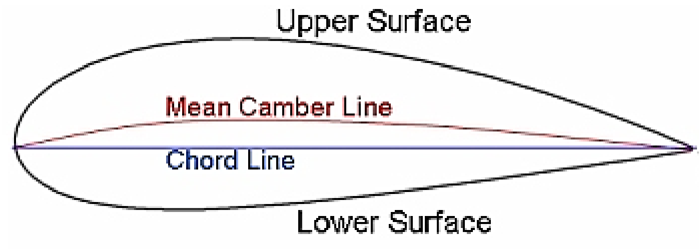




Positive camber aerofoil

If the airflow accelerates more on the upper surface that the lower surface of the aerofoil at an AoA=0°, so it will have a lift capability ($C_L > 0$). This method can be used to increase the lift capability of an aerofoil. The aerofoil will have a higher C_{Lmax}, however the adverse pressure gradient will be higher and the AoA_{CRIT} will occur earlier

In this case, when drawing the chordline and the mean camberline of the aerfoil, the mean camberline will be above the chordline. The aerofoil is **asymmetrical** and said to have a **positive camber.**

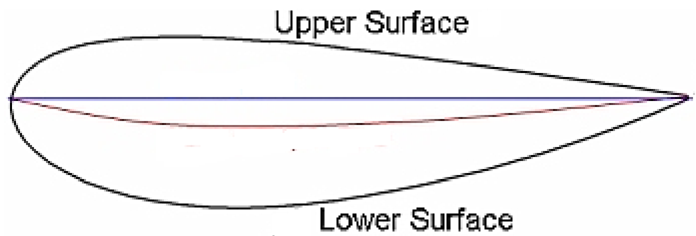


Since a positive camber aerofoil has a lift capability at AoA=0°; the AoA must be decrease below 0° for the aerofoil to have C_L=0. On the C_L curve, the curve will start behind or before the vertical axis (see graphs below)

Negative camber aerofoil

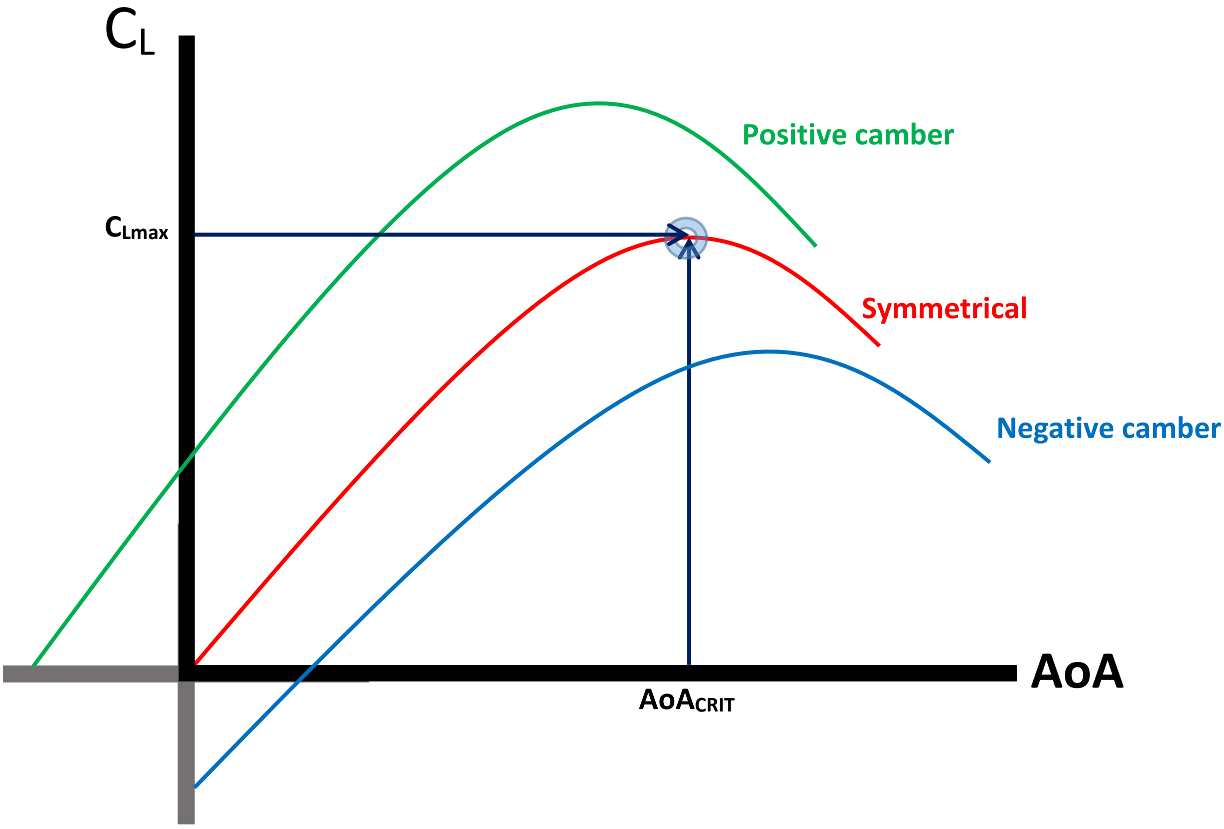
If the airflow accelerates more on the lower surface that the upper surface of the aerofoil at an AoA=0°, so it will have a <u>negative</u> lift capability ($C_L < 0$). This method can be used to generate downward lift like the aerofoil on the tailplane to stabilise the aircraft (covered later in the lesson).

In this case, when drawing the chordline and the mean camberline of the aerfoil, the mean camberline will be below the chordline. The aerofoil is **asymmetrical** and said to have a **negative camber.**



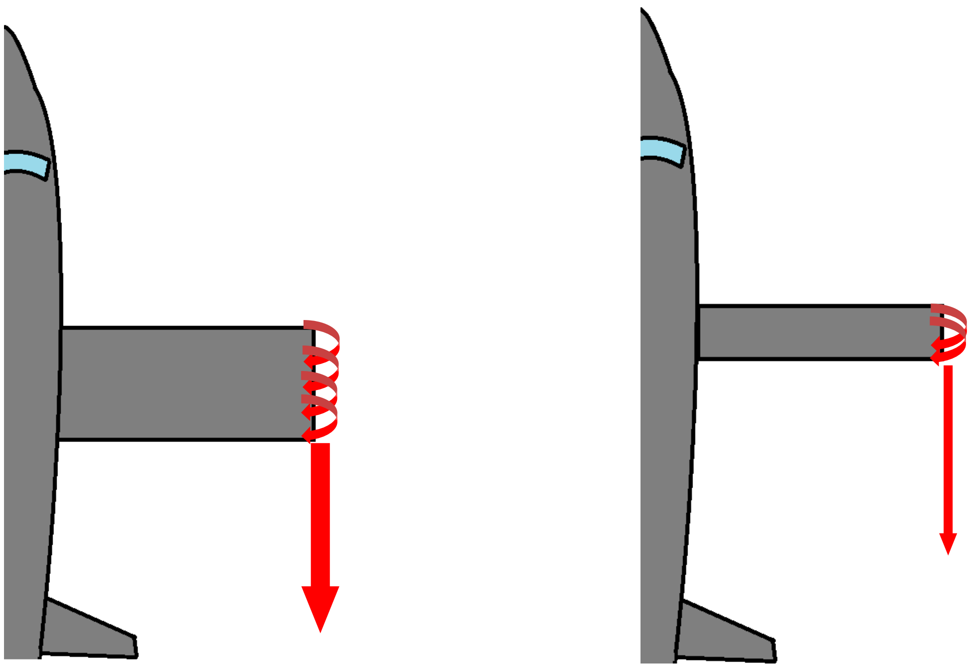
Since a zero camber aerofoil has a negative lift capability at AoA=0°; increasing the AoA will reduce the negative C_L until C_L=0, the AoA must be increased further to start to genereate positive lift capability (CL > 0). On the C_L curve, the curve will start below the horizontal axis (see graphs below)

Comparison of the Coefficient of Lift for different cambers.



4) Wing Shape

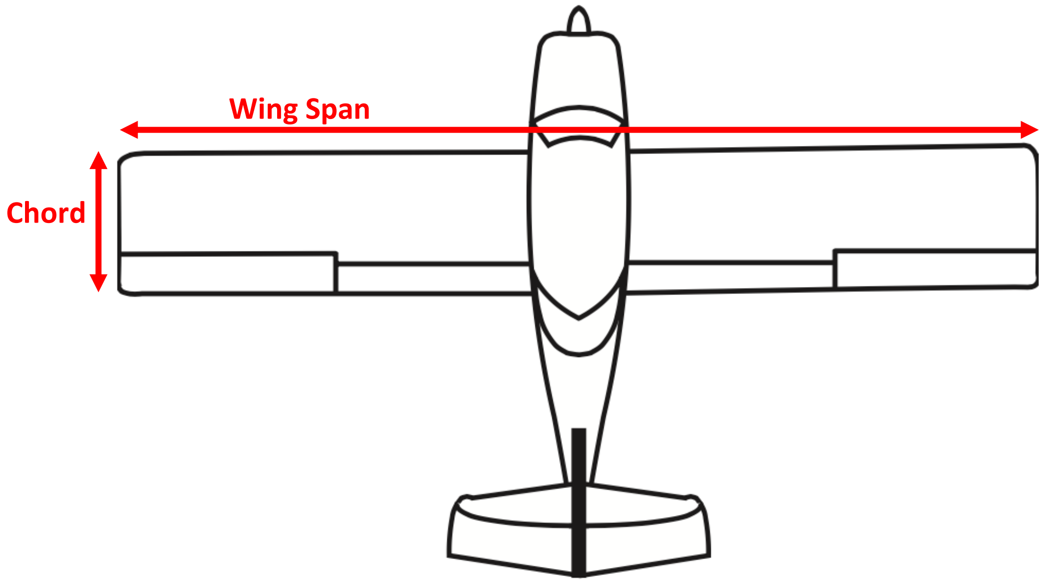
There is a method to reduce the Induced Drag, remember the Induced Drag is generated at the wing tips. So the method consist by shortening the chord, so less vortices could attach on the wing tip.



However remember the amount of Lift generated depends on the wing area

$$L = \tfrac{1}{2}\rho V^2 . \, S . \, C_L$$

The wing area is **S = Wings Span (WS) x Chordline (C)**



So if the Chord is shorten, the wing area will be smaller and so the lift generated will be less.

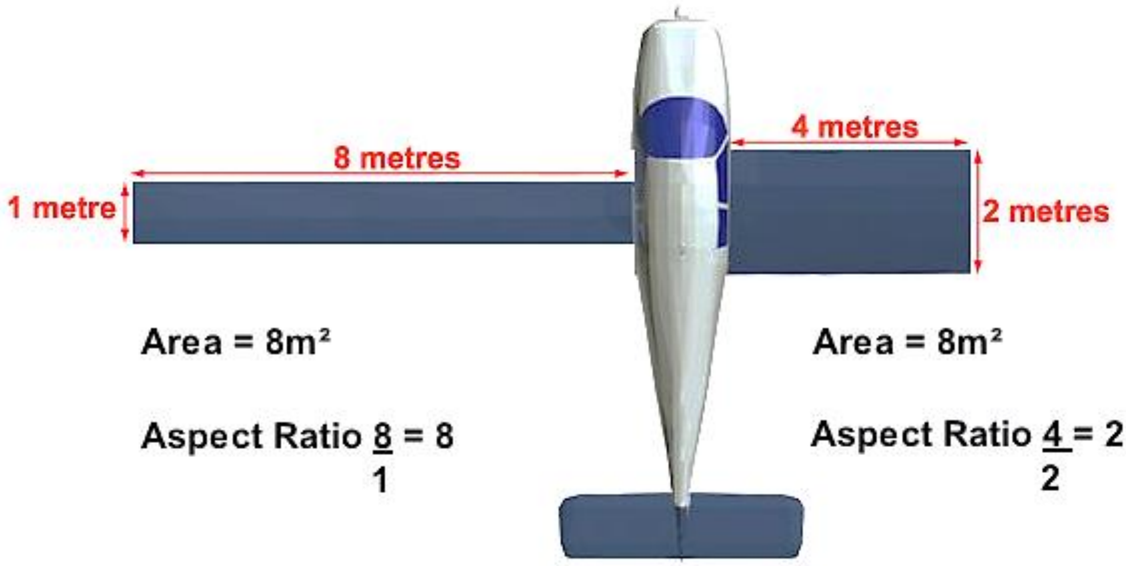
To maintain the wing area constant, the wing span could be increased.

The ratio between the wing span and the chord is called the **ASPECT RATIO (AR)**. The AR is a ratio that gives an idea about the lift capability and the Induced Drag of a wing.

$$AR = \frac{WS}{C}$$

<u>Aspect Ratio (AR)</u>

If the chord is shorten, to maintain the same wing area, the wing span is increased, however the AR increases.



An aircraft which has a high AR, means that in create low Induced Drag compared to its Lift, and an aircraft that has a low AR means that it create high Induced Drag compared to its Lift.

LOW ASPECT RATIO **HIGH ASPECT RATION**

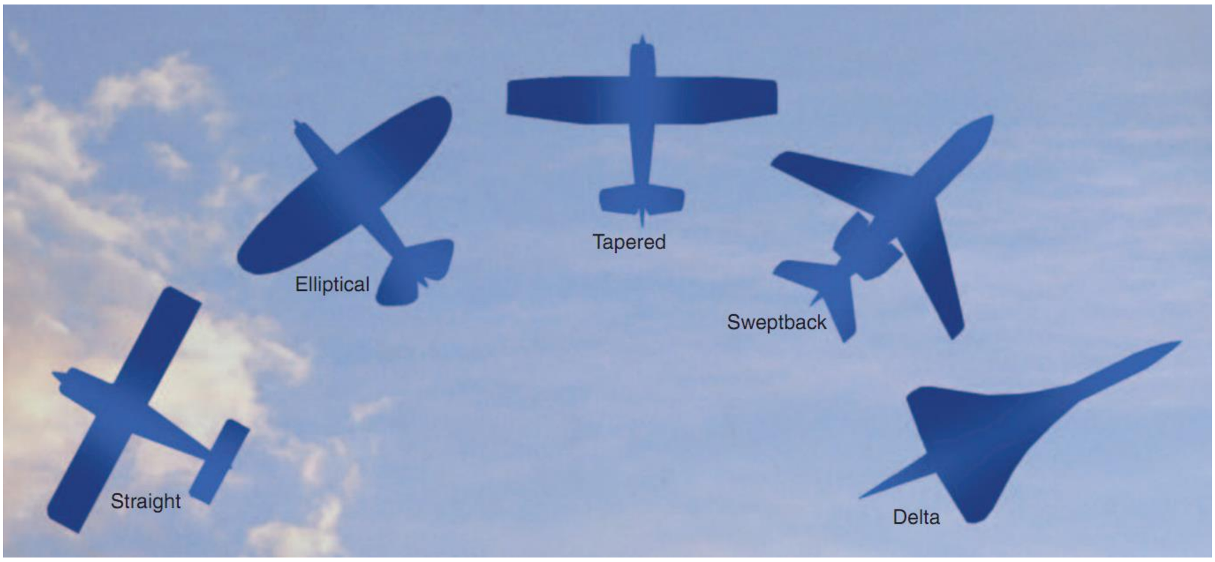
Usually the gliders (picture on the right) have a high AR, they need to minimise the drag because they cannot provide thrust, and it's only an unpowered flight.

Increasing the wingspan for a higher AR can have some inconvenient such as:

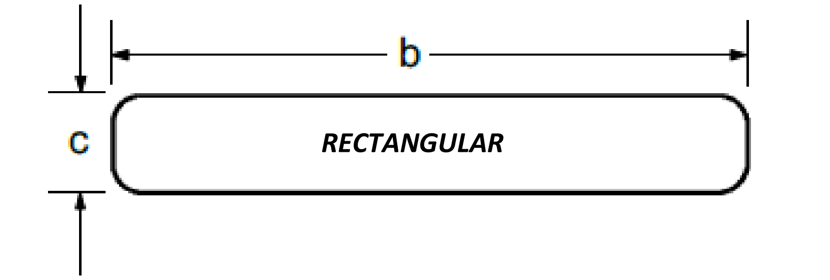
- The wings will be more subject to bending and so their stiffness and strength is reduced
- To increase the strength and stiffness of the wings, stronger spars and stronger stingers are required, so more material
- The storage and ground control could become a difficulty due to more space required for the large wingspan

Wing shapes

Other methods exist to decrease the chord without significant decrease in the wing area to minimise the Induced Drag, is simply to **shorten the Chord at the tip**. The wings is called to be tapered, but different shapes exist:



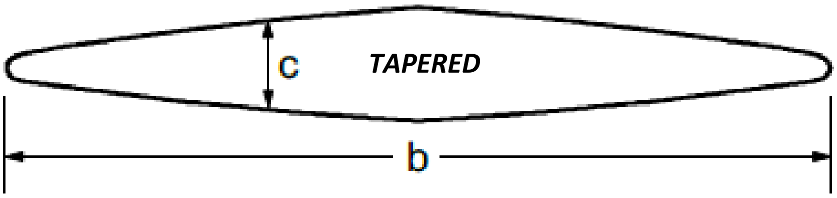
Since there are different chords on the wing, for calculation we use the **Average Chord**, which is the geometric average. The product of the span and the average chord is the wing area



b= SPAN

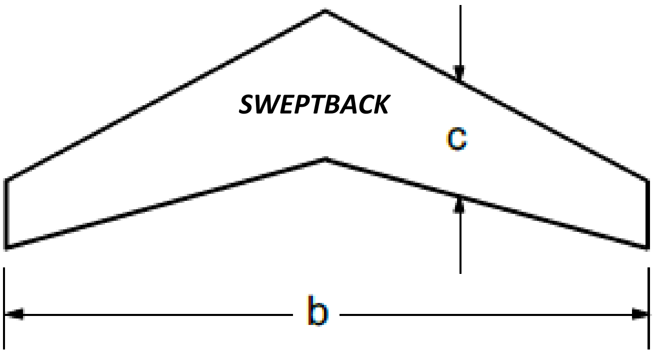
c= Average Chord

Wing Area (S) = b x c



$$\boldsymbol{AR = \frac{Span}{Average\ Chord}}$$

$$AR = \frac{\boldsymbol{b}}{\boldsymbol{c}} = \frac{b\ x\ b}{b\ x\ c} = \frac{\boldsymbol{b^2}}{\boldsymbol{S}}$$

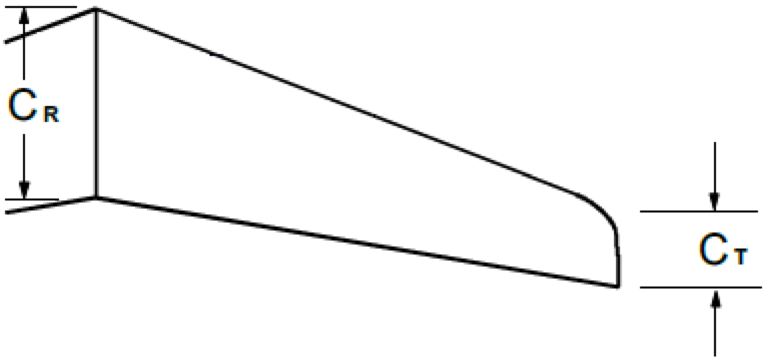


C_R= Wing Root Chord

C_T= Wing Tip Chord

Taper Ratio = C_T/C_R

Taper Ratio: The ratio of the tip chord to the root chord. The taper ratio affects the lift distribution and the structural weight of the wing. A rectangular wing has a taper ratio of 1.0 while the pointed tip delta wing has a taper ratio of 0.0

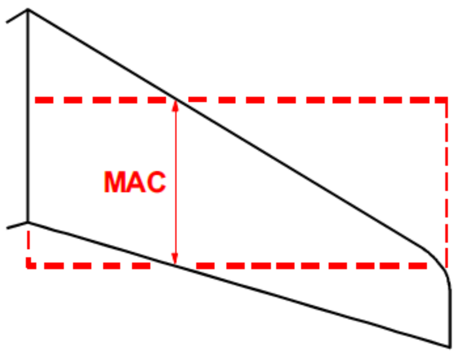


The shape of the wing will be chosen according to the operation and each shape has a different stall characteristics (covered later in the lesson)

NOTICE: The Average Chord can be confused with Mean Aerodynamic Chord

The Mean Aerodynamic Chord (MAC): The chord drawn through the geographic centre of the plan area. A rectangular wing of this chord and the same span would have broadly similar pitching moment characteristics. The MAC is located on the reference axis of the aircraft and is a primary reference for longitudinal stability considerations.

MAC = MEAN AERODYNAMIC CHORD



5) Analysis of the Induced Drag

The Induced Drag is an aerodynamic force and so can be written as the following:

$$D_i = \frac{1}{2}\rho V^2 S C_{Di}$$

This equation implies that if V increases, D_i increases, however it's the opposite, simply because C_{Di} is proportional to C_L^2 and inversely proportional to AR.

As speed increases, C_L must be reduced to keep a constant LIFT. Remember, C_L is inversely proportional to the square of the factor of the speed change.

Also, as AR increase, the D_i decrease, i.e. if the chord is twice shorter, the D_i will be twice less

$$CDi = \frac{CL^2}{AR}$$

→ If the speed is doubled

$$L = \frac{1}{2}\rho V^2 S C_L \;\rightarrow\; \frac{1}{2}\rho (2V)^2 S C_L$$

$$\frac{1}{2}\rho (2V)^2 S C_L = \frac{1}{2}\rho\, 4\, (V)^2 S C_L$$

Now the Lift is 4 times greater, so the AoA must be lower to have 4 times less C_L

$$4L = \frac{1}{2}\rho\, 4\, (V)^2 S C_L \;\rightarrow\; L = \frac{1}{2}\rho\, 4\, (V)^2 S \tfrac{1}{4}C_L$$

Then let's look at the C_{Di}

$$C_{Di} = [C_L^2/AR] \;\rightarrow\; [(\tfrac{1}{4}C_L)^2/ AR] = [\tfrac{1}{16}(C_L)^2/AR] = \boxed{\tfrac{1}{16} C_{Di}}$$

So if the speed is doubled, C_{Di} is 16 times less, if we have a look at the D_i with the speed double we have

$$D_i = \frac{1}{2}\rho V^2 S C_{Di} \;\rightarrow\; \frac{1}{2}\rho (2V)^2 S \frac{1}{16} C_{Di} = \frac{1}{2}\rho\, 4 (V)^2 S \frac{1}{16} C_{Di}$$

$$= \frac{1}{2}\rho\, 4\, (V)^2 S \tfrac{1}{4}x\tfrac{1}{4}C_{Di} = \frac{1}{2}\rho V^2 S \tfrac{1}{4}C_{Di} = \boxed{\tfrac{1}{4}D_i}$$

If speed is doubled/halved, C_{Di} will be (1:16)/16 times of its previous value and Di is ¼/4 times of its previous value.

C_{Di} is inversely proportional to the square of the factor of the speed change

C_{Di} = (Old Speed / New Speed)^2