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AERODYNAMICS 1

CHAPTER 3 – LIFT

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THE LIFT FORMULA

Introduction

The Lift formula is derived from the experiments carried out by Bernoulli, and the equation of continuity. This formula simply states that the total lift produced by an aerofoil is dependant on the values of other factors, these factors being:

- **Free Stream Velocity or Relative Airflow (V^2)**
- **Air Density (ρ)**
- **Wing Area (S)**
- **Coefficient of Lift (C_L) which includes the following, as well as any unknown quantities that are not otherwise represented in the formula:**
    - **Wing Shape in Section and Planform**
    - **Angle of Attack**
    - **Condition of the Surface**
    - **Viscosity of the Air**
    - **The Speed of Sound**

The Lift Formula is therefore represented by:

$$\text{Lift} = C_L \tfrac{1}{2} \rho V^2 S$$

Therefore any changes in any of these factors will affect the total Lift.

CONTROLLING LIFT

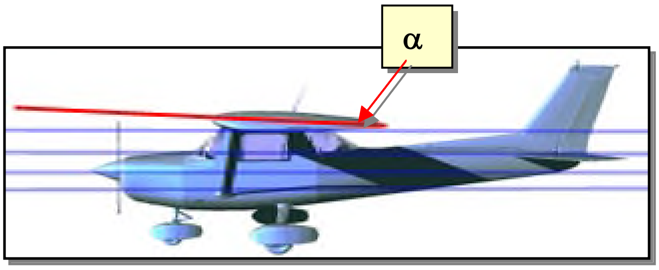
INTRODUCTION

Although an aircraft wing is designed to develop lift at the best possible rate for the aircraft design, the pilot has a requirement to be able to increase or decrease the amount of lift being developed by the wing. In the design of the aircraft, controls and techniques are therefore introduced to provide the pilot with the tools to control the lift produced. Some of these tools include:

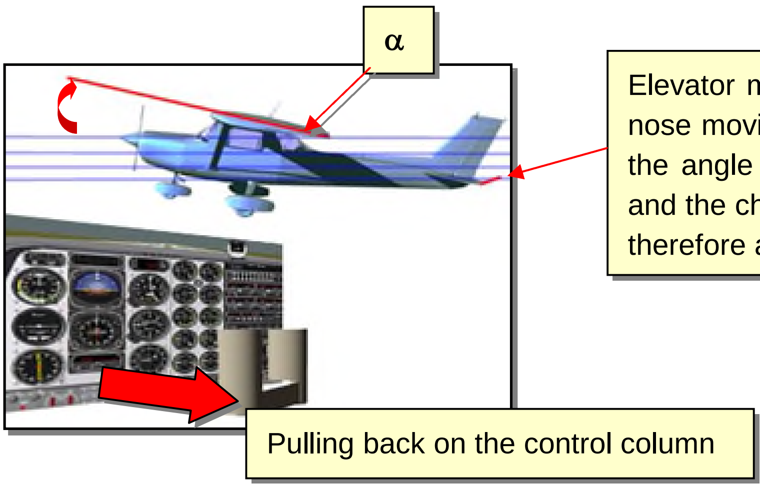
- ***Control of the angle of attack***
- ***Control of airspeed***
- ***Changing the shape of the wing (using Flaps)***

CHANGING THE ANGLE OF ATTACK (α)

α = The angle between the relative airflow and the chord line



The pilot has direct control of the angle of attack of the aircraft. By moving the control column forward or backward, the elevators are deflected. This deflection during normal speed operations, alters the angle of attack of the aircraft.



Elevator moves up, which results in the nose moving up, which in turn results in the angle between the relative air flow, and the chord line (α) increasing. This is therefore a greater angle of attack.

These changes in the angle of attack (α), also result in changes in the ***coefficient of lift***.

THE COEFFICIENT OF LIFT (C_L)

The coefficient of lift is a way of measuring lift as it relates to the angle of attack, and is measured and determined in a wind tunnel. ***The wing design and conditions as well as the angle of attack will form the basis of C_L .***

From the Lift Formulae

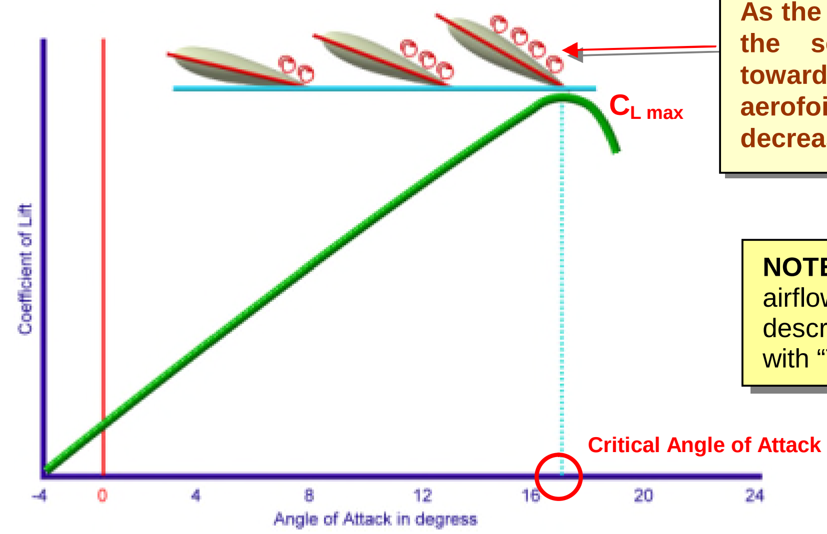
$$L = CL \ \tfrac{1}{2} \rho V^2 S$$

We know that by increasing the value of C_L, we will increase the value of total lift.

We are now faced with the question : ***"Will the C_L and therefore Lift continue to rise as the angle of attack is increased indefinately ?"***

The answer to this question is a definite no !

Each aircraft has a point at which maximum lift is produced by the aerofoil (C_{Lmax}). This angle is called the critical angle of attack, or the stalling angle. Beyond this point, lift instead of increasing, decreases as the aerofoil stalls.



As the angle of attack increases, the separation point moves toward the leading edge of the aerofoil, with a resulting decrease in C_L

NOTE: Separation of laminar airflow into turbulent airflow is described in the section dealing with "The Stall".

Although the pilot has control of producing more or less lift by manipulating the angle of attack, this is limited to those angles of attack within the aircraft's parameters. ***An aircraft will always stall at a specific angle of attack, irrespective of the speed or weight of the aircraft***.

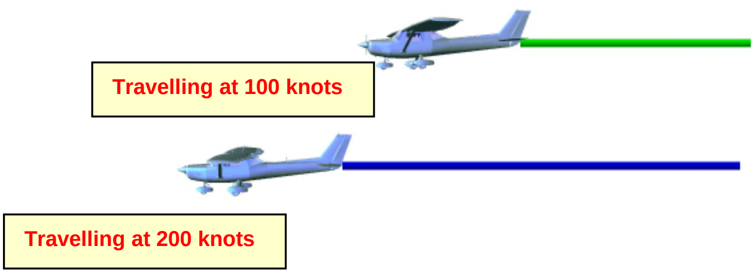
Once the aircraft reaches the stall, the pilot will feel a buffet of the controls, as a result of the turbulence over the control surfaces. In order to restore the smooth airflow, the angle of attack must be reduced.

CHANGING THE SPEED

From the Lift formulae, the relationship between lift and speed can be seen. This relationship is simply, that the faster the wing moves through the air, the more lift it will produce.

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

By way of an example, let us take two identical aircraft, with the same parameters, but with different speeds.



$L = 1\ \tfrac{1}{2}\ 1\ 100^2\ 1$
L=5000

$L = 1\ \tfrac{1}{2}\ 1\ 200^2\ 1$
L=20000

The values in the lift formulae are not true values. They are merely to indicate the relationship between lift and speed.

From this example, we can see that the lift produced by an aircraft travelling twice the speed is four times more than that for the other aircraft.

Therefore if the faster aircraft were to decrease it speed by half (100knots), its lift would reduce to a ¼ of its original value.

Depending on the aircraft type, configuration and weight, a specific airspeed may be required to be maintained in order for the aircraft to maintain straight and level flight.

THE RELATIONSHIP BETWEEN ANGLE OF ATTACK AND AIRSPEED

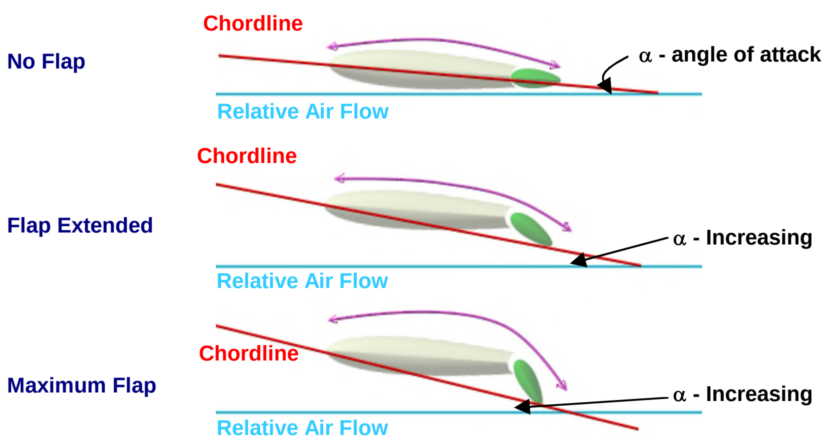
$$L = C_L \tfrac{1}{2} \rho V^2 S$$

From the lift formulae, we know that as angle of attack increases (and therefore C_L) total lift increases. We also know that as V^2 increases, total lift also increases. In order therefore to maintain a given amount of lift, the pilot will have to decrease C_L (angle of attack), if speed were to increase. If speed were to decrease the pilot would have to increase C_L (angle of attack) to maintain the required lift.

USING FLAP TO CONTROL LIFT

The correct and efficient use of flaps increases the lifting cabability of the aerofoil. What happens to the aerofoil when flaps are lowered?

In order to answer this, we must look at an aerofoil of an aircraft in level flight, at a given speed.



1. As flaps are lowered, the chordline changes (straight line between the leading edge and the trailing edge).

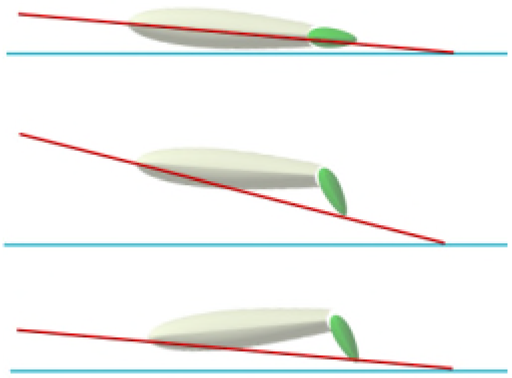
This change in chordline, means that the angle (angle of attack) between the chordline and the relative air flow increases. The increase in angle of attack, means an increase in C_L.

2. As flaps are lowered, camber of the wing is increased. Increased camber means an increase in the C_L

Therefore in order to maintain the required lift to keep the aircraft straight and level, the pilot will have to reduce the angle of attack by lowering the nose of the aircraft.

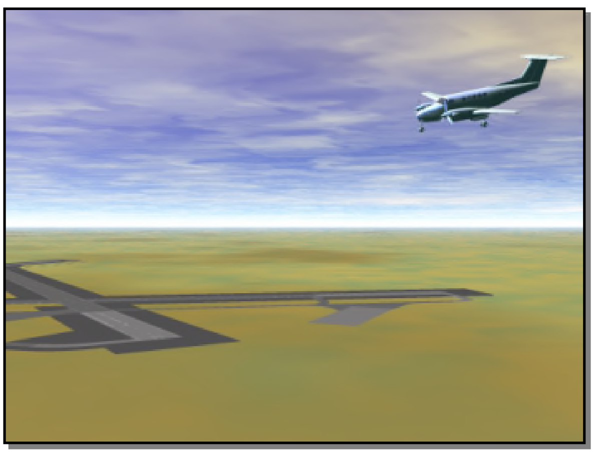
THE ATTITUDE OF THE AIRCRAFT

The lowering of the flaps changes the attitude of the aircraft. Remember that to compensate for the increased angle of attack, the nose is lowered. This lowering of the nose restores the angle of attack to that required for level flight, but changes the attitude of the aircraft.



Flaps lowered - α increase

Nose lowered - α decreased



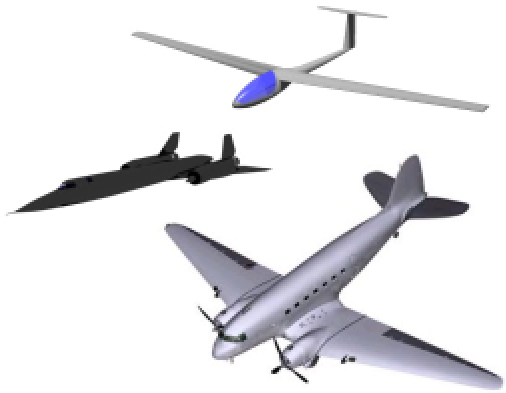
Although the lowering of flaps increases the lift of the aerofoil, it also increases drag. This increase in drag, means that the pilot will have to increase the thrust of the aircraft in order to maintain a required speed in level flight, lower the nose further with no additional thrust to maintain speed which will result in a descent. This is typical of a landing aircraft.

With a departing aircraft, flaps are lowered for the take-off. This allows the aircraft to take-off with a lower speed.

DESIGN FACTORS AND LIFT

INTRODUCTION

Lift is an essential part of aerodynamic design, and not all aircraft will require the same lift characteristics.



Different design features incorporated in an aircraft will result in various possibilities of lift creation, and these features can be utilised as required for the proposed role of the aircraft.

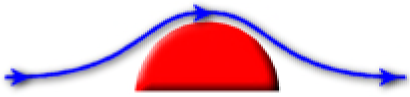
This study will deal with the various structural design factors, and the effect they have on the amount of lift created.

SHAPE OF WING AND PLANFORMS

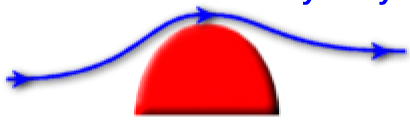
LEADING EDGE (LE) RADIUS

According to Bernoulli's Theorem, when air flows over a curved surface, its velocity will increase.

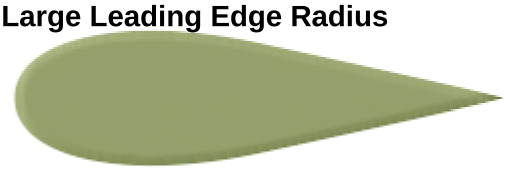
For a given free stream velocity, the higher the curvature of the surface, the greater the acceleration will be.

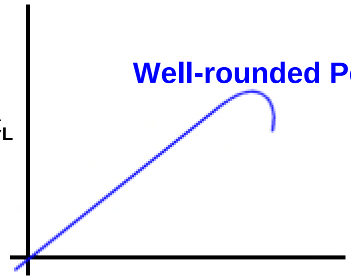


Conversely, as the radius of the curvature of the surface decreases, the airflow will find it increasingly more difficult to follow the contour of the surface, and will tend to break away from the surface.



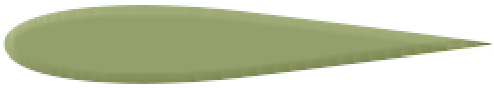
The shape of the leading edge (LE) and condition of its surface largely determines the stalling characteristics of a wing.

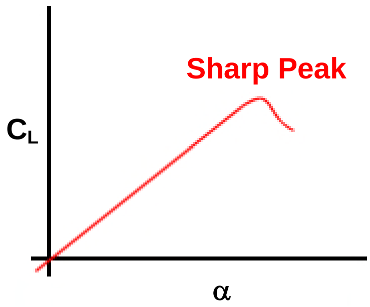




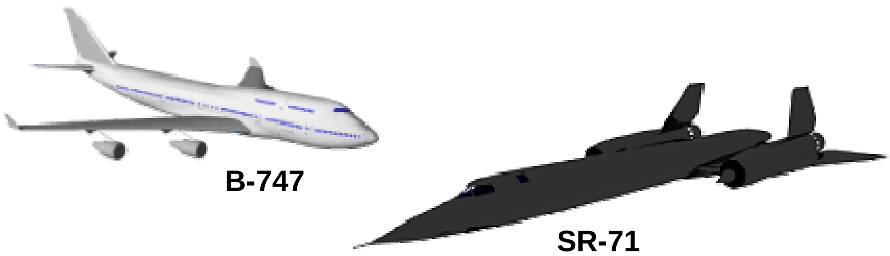
In general a blunt LE with a large radius will result in a well-rounded peak to the C_L curve.

Small Leading Edge Radius



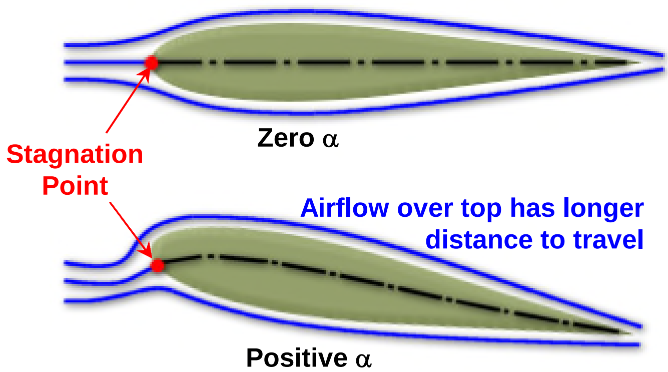


In contrast, a sharp LE will have a small radius and the sharp peak of the C_L curve will result in an abrupt stall.



In the image above, two aircraft are shown. The Boeing 747 on the left has a Large Leading Edge Radius, whilst the SR-71 on the right has a Small Leading Edge Radius.

CAMBER



A symmetrical wing at zero angle of attack will have the same pressure distribution on its upper and lower surface and will therefore not produce lift since they cancel each other out.

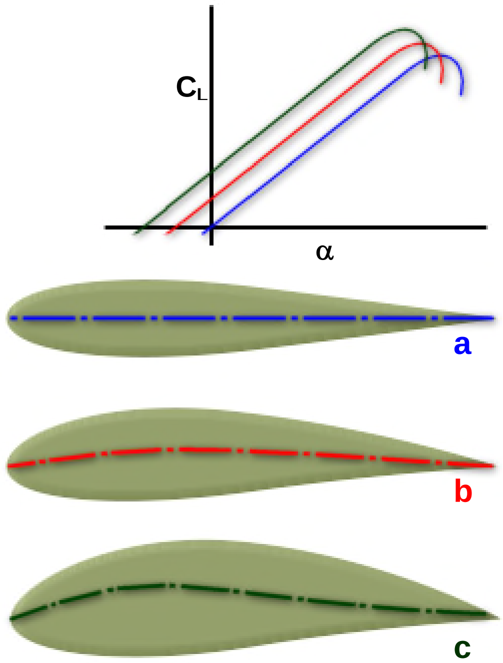
As the angle of attack increases the **stagnation point** moves from the chord line to a point below the centre of the LE.

This effectively lengthens the path of the flow over the top surface and reduces it on the lower surface thus changing the symmetrical section to an apparently cambered surface.

Stagnation Point

Stagnation point is the centre of the high pressure area ahead of the aerofoil.

Positive camber produces lift at zero angle of attack because the airflow attains a higher velocity over the upper surface creating a pressure differential and therefore Lift.



The effect of camber on C_L can be illustrated by the following graph.

Line (**a**) represents the curve for a symmetrical section.

Lines (**b**) and (**c**) are for sections of increasing camber.

Aerofoil (**b**) and (**c**) have an advantage over symmetrical aerofoils at all angles of attack but pay the penalty of an earlier stalling angle as is shown by C_L versus angle of attack, **which shifts up and left**.

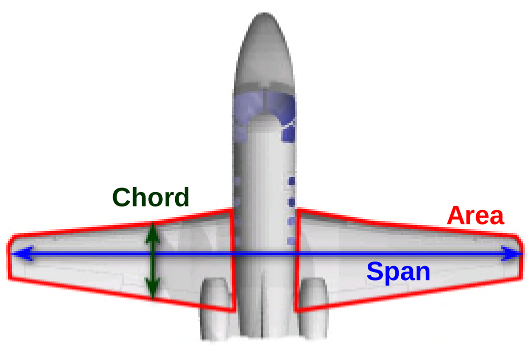
ASPECT RATIO

The formula for Aspect Ratio is:

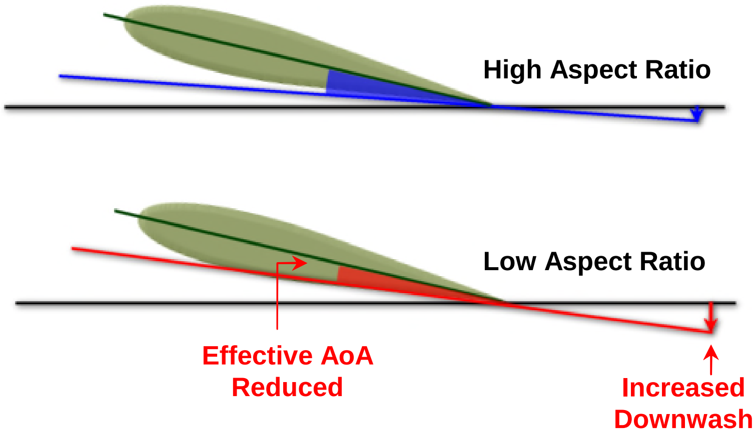
$$\mathbf{A} = \frac{\text{Span}^2}{\text{Area}}$$

or

$$\mathbf{A} = \frac{\text{Span}}{\text{Average Chord}}$$



NOTE: The downwash associated with wingtip vortices is more fully described in the section dealing with Induced Drag (Vortex Drag)



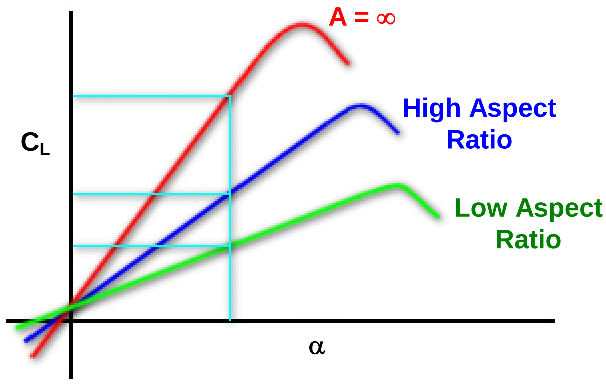
Any wing has wingtip vortices that reinforce the downwash that reduces the effective angle of attack.

A low aspect ratio wing (such as the wing of the F104) produces more downwash than a high aspect ratio wing due to the wingtip vortices being bigger.

Therefore the effective angle of attack (Coefficient of Lift) is reduced more than with a wing of higher aspect ratio.

On the graph the **red line** indicates a wing of infinite span; the **blue line** a finite wing of high aspect ratio and the **green line** a finite wing of low aspect ratio.

The **cyan lines** give an indication of the difference in Coefficient of Lift with the three wings at the same angle of attack.



SWEEPBACK

A good way to demonstrate the effect of sweepback on the C_L is to have a look at an aircraft that has variable wings such as a F-14.

For this purpose the first formula for aspect ratio will be used. It does not matter what the position of the wings are; the wing area will always stay constant.

$$A = \frac{Span^2}{Area}$$

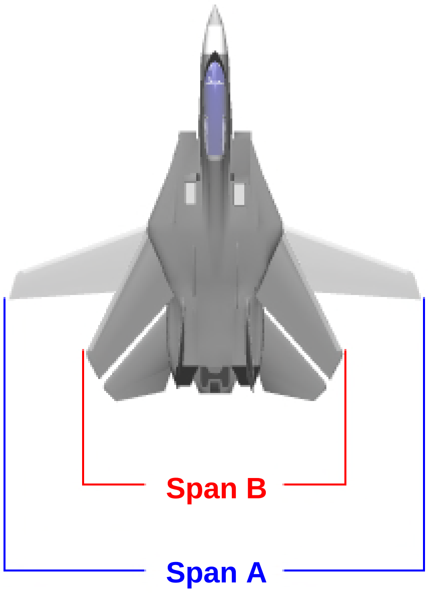
Therefore the only variable in the formula is the span, which will change as the wing sweep angle changes.

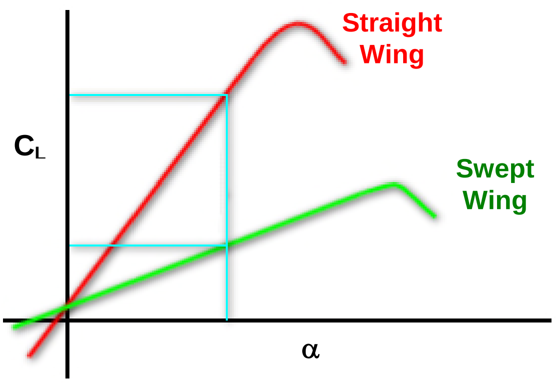
With the wings in the forward swept position the span equals A.

With the wings in the backward swept position the span equals B.

From the formula it can be seen that the Aspect Ratio for span B will be less than for span A.

Due to the decrease in Aspect Ratio with the wing swept back, there will be a decrease in the C_L.

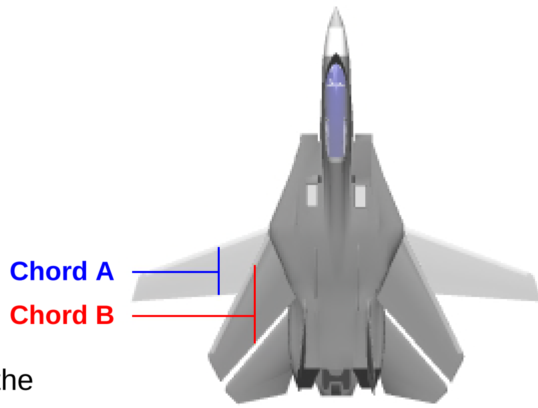




From the graph the effect of sweepback on C_L at a constant angle of attack can be seen.

Using the second formula for aspect ratio has the same result.

$$A = \frac{\text{Span}}{\text{Average Chord}}$$

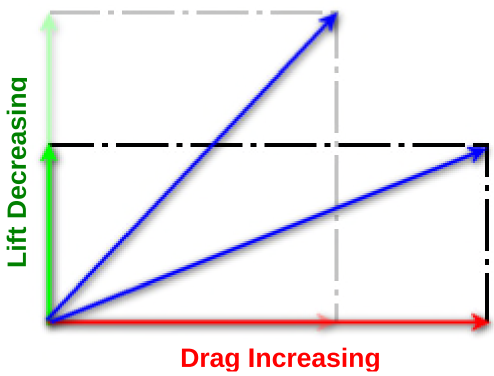


It is easy to see that as the wings are swept back **(B)**, the chord of the wing effectively increases as the span reduces, thus reducing the value of the formula and the aspect ratio.

CONDITION OF WING SURFACE

Surface irregularities cause the laminar part of the boundary layer to become turbulent. This increases surface friction and therefore Drag.

As the **Drag** increases, the vector representing the **total reaction** tilts to the right.



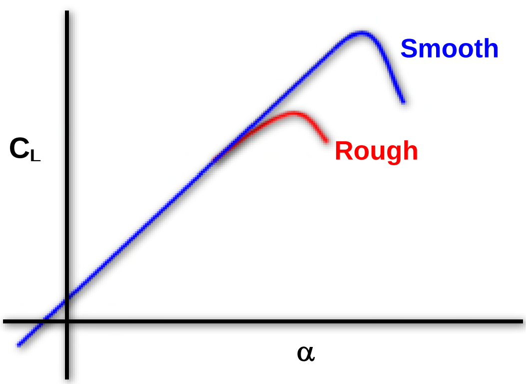
A corresponding decrease in **Lift** occurs.

$$\text{Lift} = C_L \tfrac{1}{2}\rho V^2 S$$

In the Lift formula density, velocity and surface area stay constant. Therefore for the equation to be true **C_L must decrease** with a decrease in Lift.

The maximum Coefficient of Lift is very sensitive to any surface roughness, especially on the leading edge and the first 20% of the chord length.

In general the maximum lift coefficient decreases progressively for the same angle of attack with an increase in surface roughness due to mud, ice and combat damage.



AEROFOILS

Aerofoil performance is determined by its contour. Generally, aerofoils can be divided into three classes:

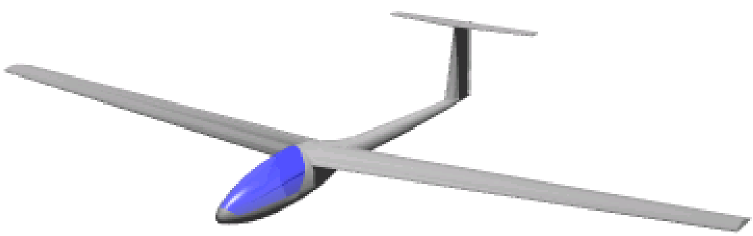
- High Lift Aerofoils
- General Purpose Aerofoils
- High Speed Aerofoils

HIGH LIFT AEROFOILS

High lift aerofoils are mainly used on sailplanes and aircraft where a high C_L is important and speed is a secondary consideration.

Typical characteristics of high lift aerofoils are:

- A high thickness to chord (T/C) ratio.
- A pronounced camber.
- A well - rounded leading edge.
- Maximum thickness is at about 25% - 30% of the chord aft of the leading edge.



The greater the camber; the greater the shift of centre of pressure for a given change in angle of attack.

This leads to a large range of movement of the centre of pressure on a high lift section.

GENERAL PURPOSE AEROFOILS

General purpose aerofoils are used on aircraft whose duties require speeds that are not high enough to subject the aerofoil to the effects of compressibility. An example of such an aircraft is the DC - 3.

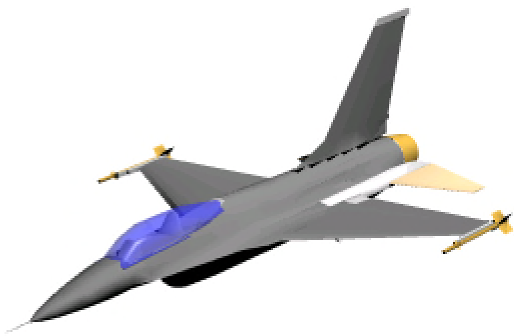
Typical characteristics of general-purpose aerofoils are:

- A lower t/c ratio
- Less camber
- A sharper leading edge
- Maximum thickness is still at about 25% - 30% of the chord aft of the leading edge.

The lower t/c ratio results in less drag and a lower C_L than those of a high lift aerofoil.

HIGH SPEED AEROFOILS

These aerofoils are usually symmetrical about the chord line. Some sections are wedge shaped whilst others consist of arcs of a circle placed symmetrically about the chord line. An example of this type of aerofoil is used on the F - 16.



Typical characteristics of high speed aerofoils are:

- A very low t/c ratio
- No camber
- A sharp leading edge
- Maximum thickness is at about 50% chord point

Most of these sections lie in the 5% - 10% t/c ratio band. Some research aircraft have even thinner sections due to the over-riding requirement for low drag.