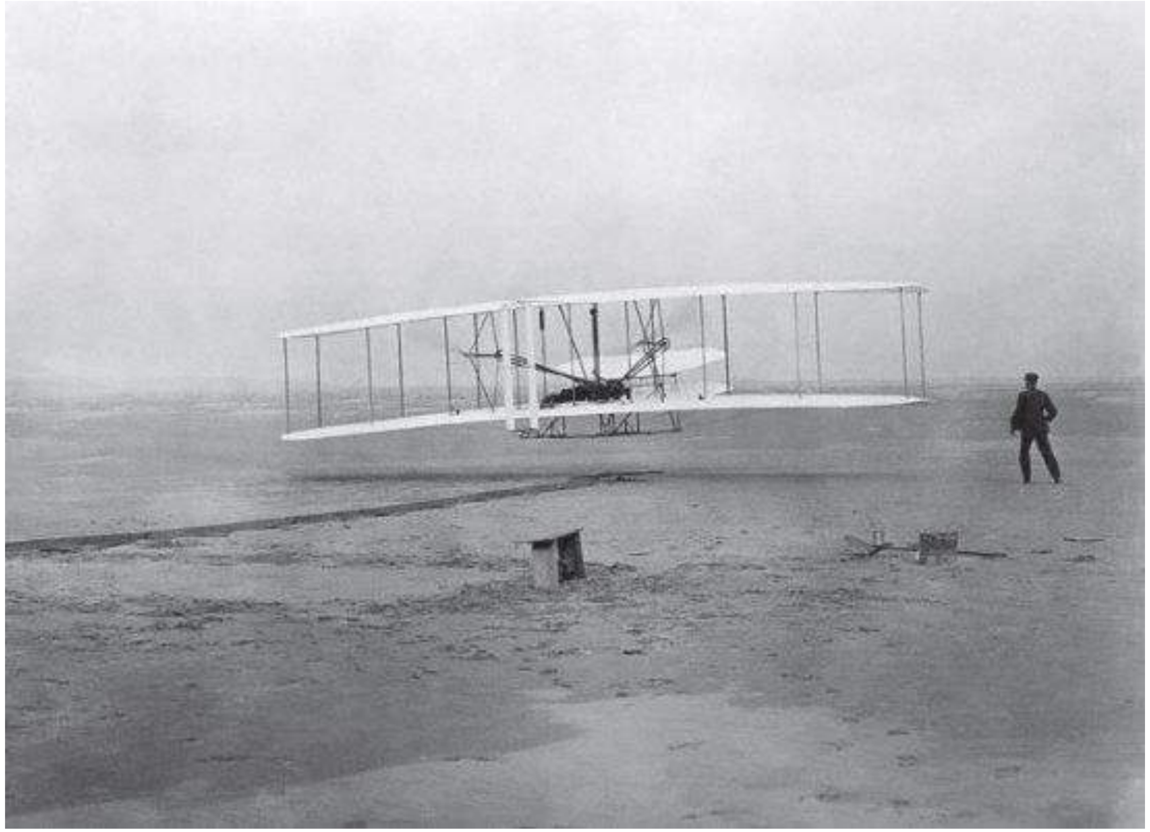
Introduction to Aircraft

- Wright Brothers (Orville and Wilbur Wright) made the historic first flight on a powered plane at Kitty Hawk, North Carolina, USA on December 17, 1903.
- Orville l



Wright Brothers laid the foundation stone for the aircraft design and development by:

1. Constructing a wind tunnel
2. Developing a comprehensive flight control system with enough control capability
3. Designing a light-weight engine and an efficient propeller

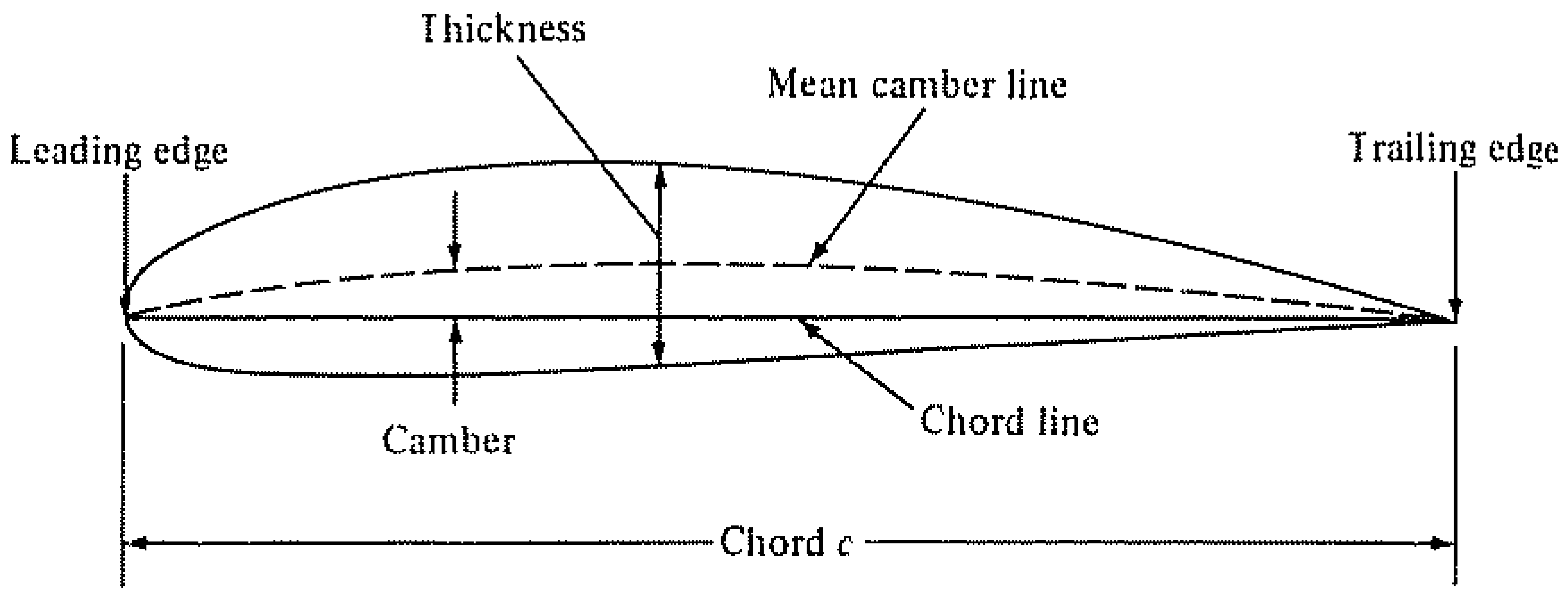
- There has been a paradigm shift of late in the present-day warfare, namely deployment of Unmanned Air Vehicles—UAVs—or also known as Drones.
- These UAVs are flown safely by “pilots” situated several thousands of miles away, and they are far less expensive than manned fighters.
- The UAV “pilots” also have a pseudo cockpit with similar instrument layout.

Aircraft

1. Main frame—fuselage to carry passengers or payloads,
2. Wings to provide lifting force to overcome weight of the aircraft
3. Propulsion system (jet engine or turboprop or propeller engine) and
4. Sophisticated avionics system including instrumentation system, navigation systems, communication systems and warning systems.

Introduction on airfoils

AIRFOIL NOMENCLATURE



- **Mean Chamber Line:** Set of points halfway between upper and lower surfaces
- **Leading Edge:** Most forward point of mean chamber line
- **Trailing Edge:** Most reward point of mean chamber line
- **Chord Line:** Straight line connecting the leading and trailing edges
- **Chord, c:** Distance along the chord line from leading to trailing edge
- **Chamber:** Maximum distance between mean chamber line and chord line

HOW DOES AN AIRFOIL GENERATE LIFT?

1. Flow velocity over top of airfoil is faster than over bottom surface

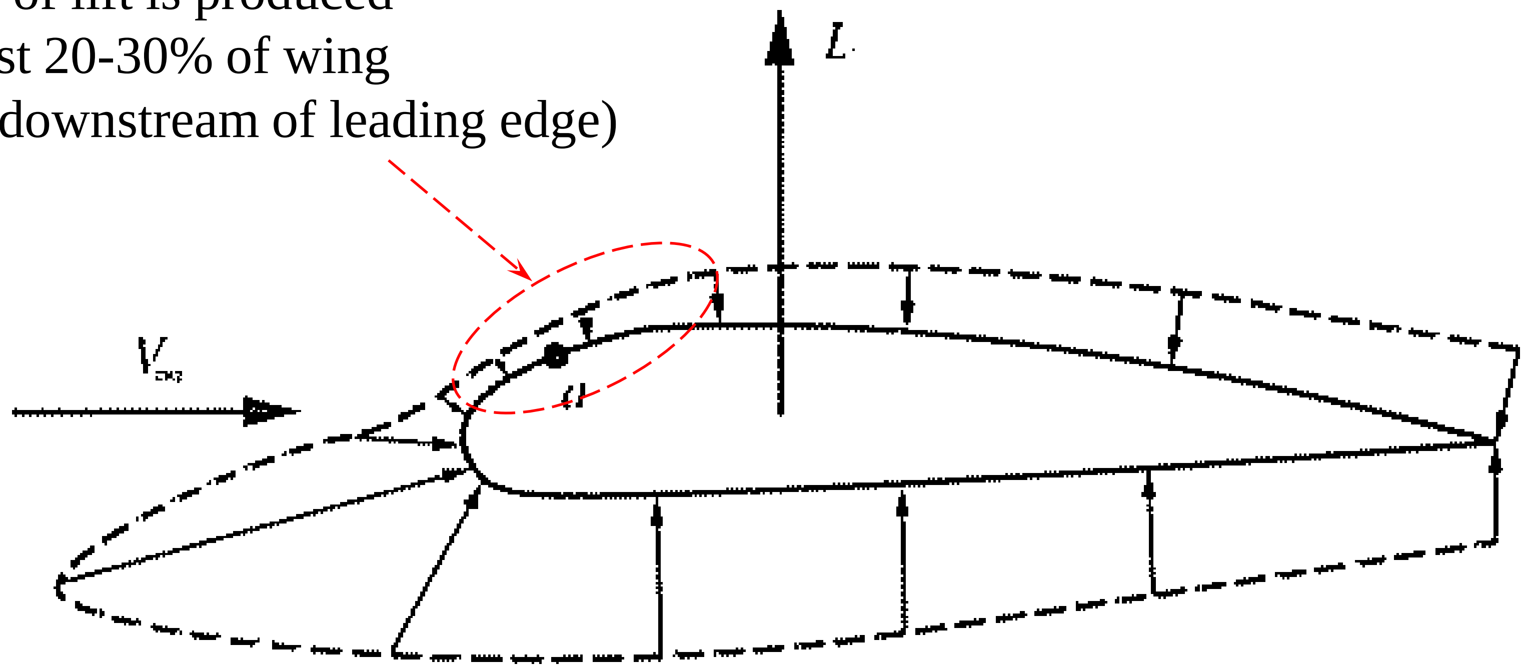
2. As V ↑ p↓
   - Bernoulli's Equation

$$p + \frac{1}{2}\rho V^2 = \text{constant}$$

   - Called **Bernoulli Effect**

3. With lower pressure over upper surface and higher pressure over bottom surface, airfoil feels a net force in upward direction → Lift

Most of lift is produced
in first 20-30% of wing
(just downstream of leading edge)



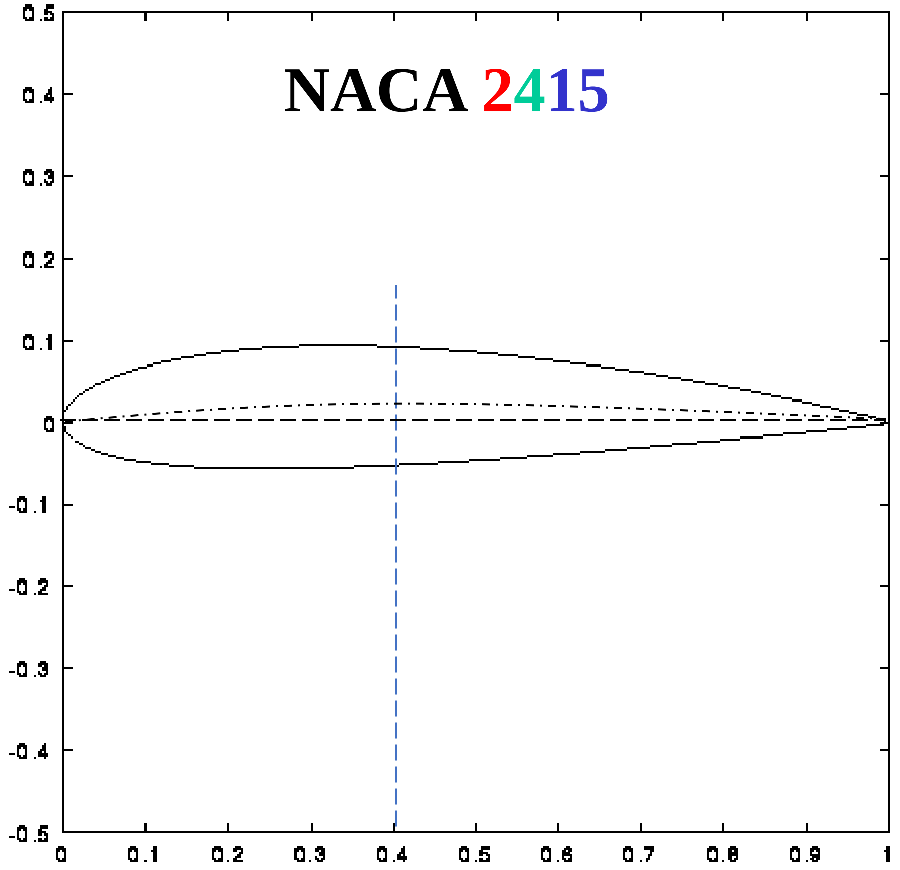
NACA FOUR-DIGIT SERIES

- **First digit** specifies maximum camber in percentage of chord
- **Second digit** indicates position of maximum camber in tenths of chord
- **Last two digits** provide maximum thickness of airfoil in percentage of chord

- National Advisory Committee for Aeronautics (NACA)

Example: **NACA 2415**

- Airfoil has maximum thickness of **15%** of chord (0.15c)
- Camber of **2%** (0.02c) located **40%** back from airfoil leading edge (0.4c)



WHAT CREATES AERODYNAMIC FORCES?

- Aerodynamic forces exerted by airflow comes from only two sources:

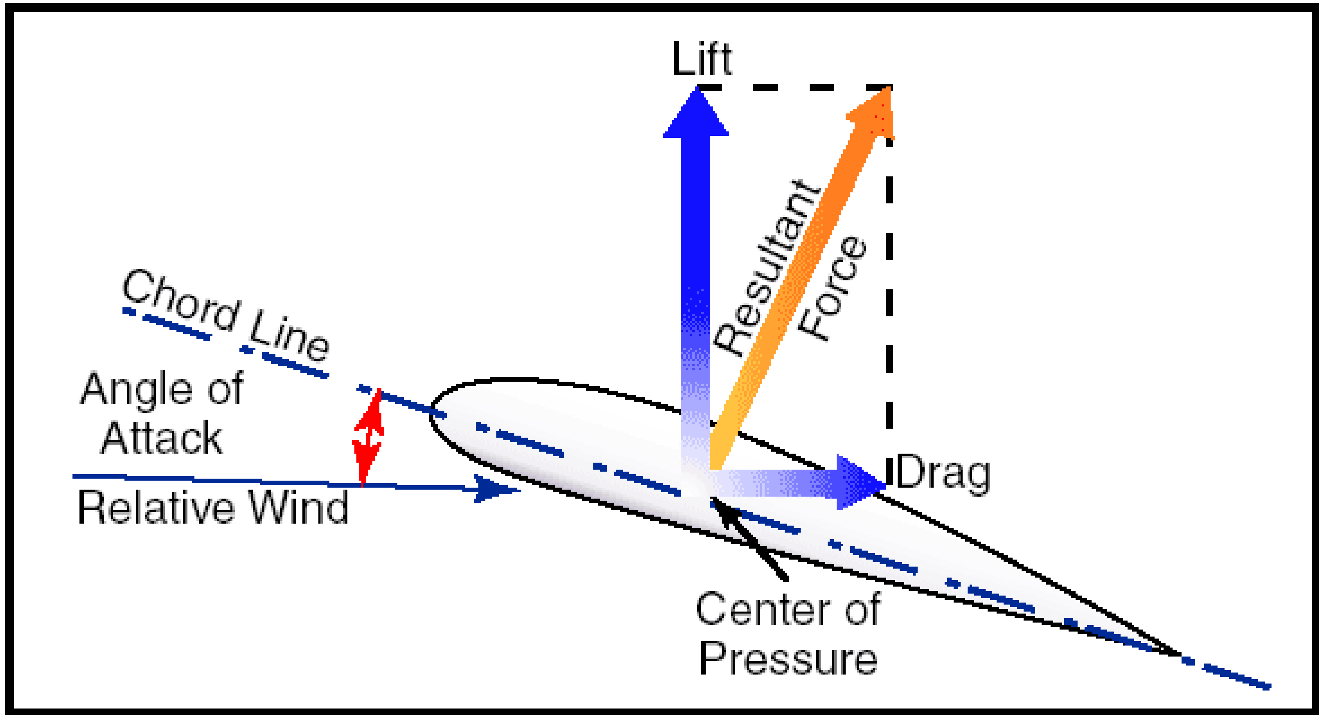
1. Pressure, p, distribution on surface
   - Acts normal to surface
2. Shear stress, t_w, (friction) on surface
   - Acts tangentially to surface

- Pressure and shear are in units of force per unit area (N/m^2)
- Net unbalance creates an aerodynamic force

RESOLVING THE AERODYNAMIC FORCE

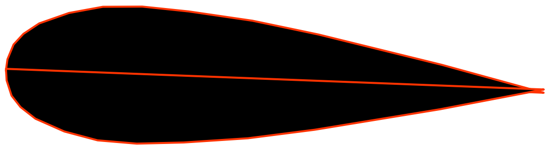
- **Relative Wind:** Direction of V_∞
- **Angle of Attack, α:** Angle between relative wind (V_∞) and chord line

- Total aerodynamic force, **R**, can be resolved into two force components
  - **Lift, L:** Component of aerodynamic force **perpendicular** to relative wind
  - **Drag, D:** Component of aerodynamic force **parallel** to relative wind



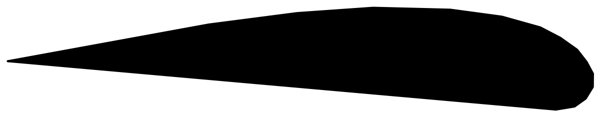
Types of Airfoils

a)Symmetrical



- Equal chamber on each side
- Each half mirror image of other
- Mean chamber line and chord line are coincidental
- Produces zero lift at zero angle of attack

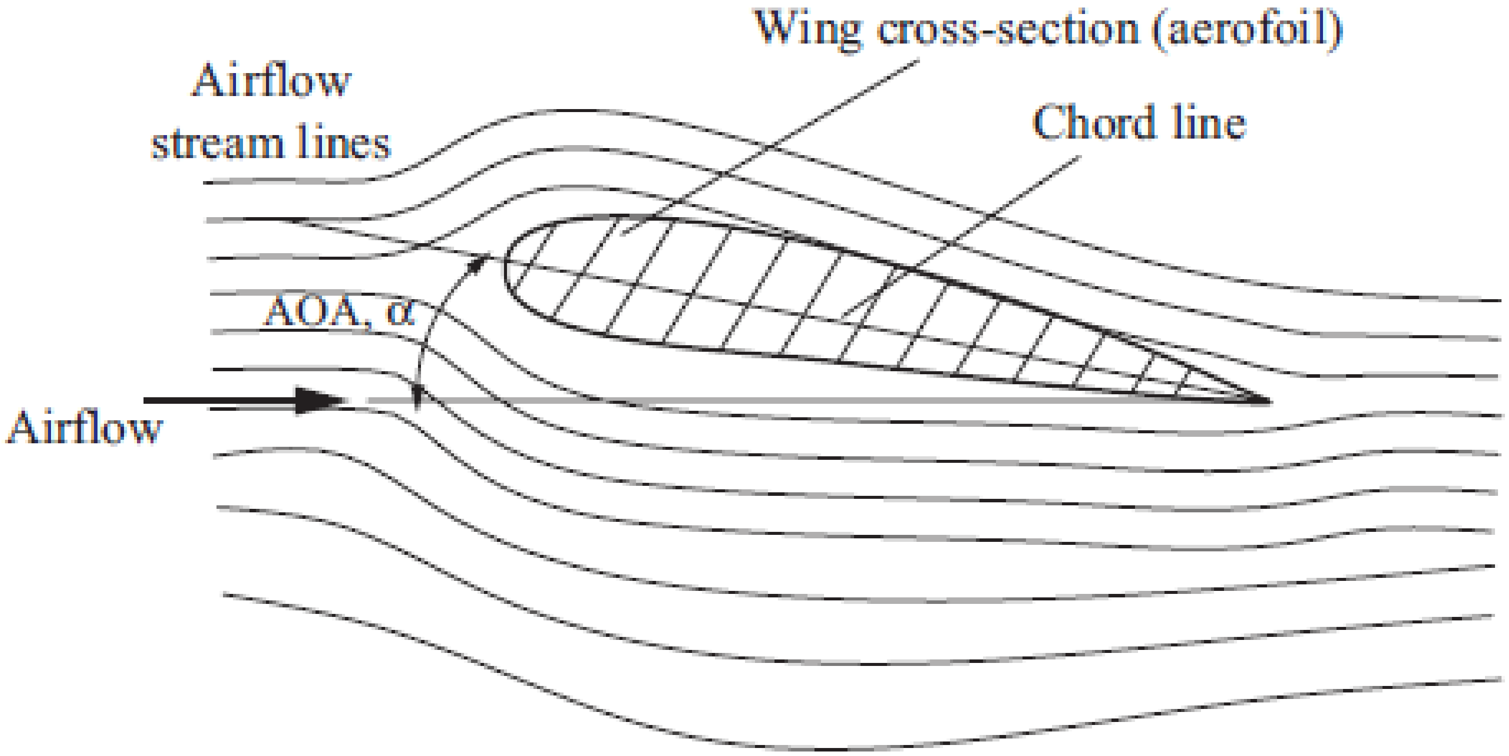
b)Nonsymmetrical (Chambered)



- Greater curvature above the chord line than below
- Chord and chamber lines are not coincidental
- Produces useful lift even at negative angles of attack
- Produces more lift at a given angle of attack than symmetrical
- Better stall characteristics than symmetrical
- Good lift to drag ratio
- Limited to low relative wind velocity, <300 knots

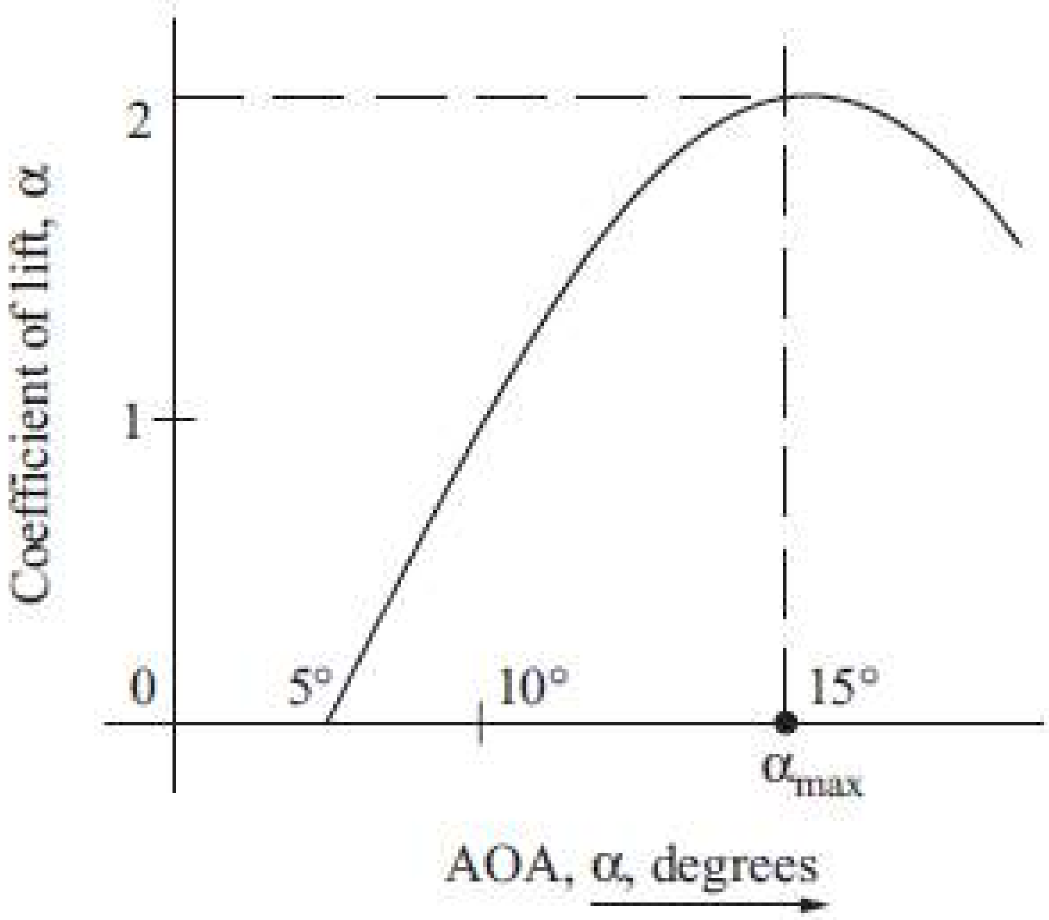
Angle of Attack (AOA)

- Angle of attack (α) is the angle between the chord line of an aerofoil and the vector representing the relative motion of aerofoil and the surrounding air.



There is another related angle, called pitch angle which is different from angle of attack—pitch angle is measured with respect to the horizon, whereas AOA is measured with respect to the direction of local airflow.

- The lift coefficient, *CL* of a fixed-wing aircraft is directly related to AOA. Increasing α increases *CL* up to the maximum lift, after which lift decreases.



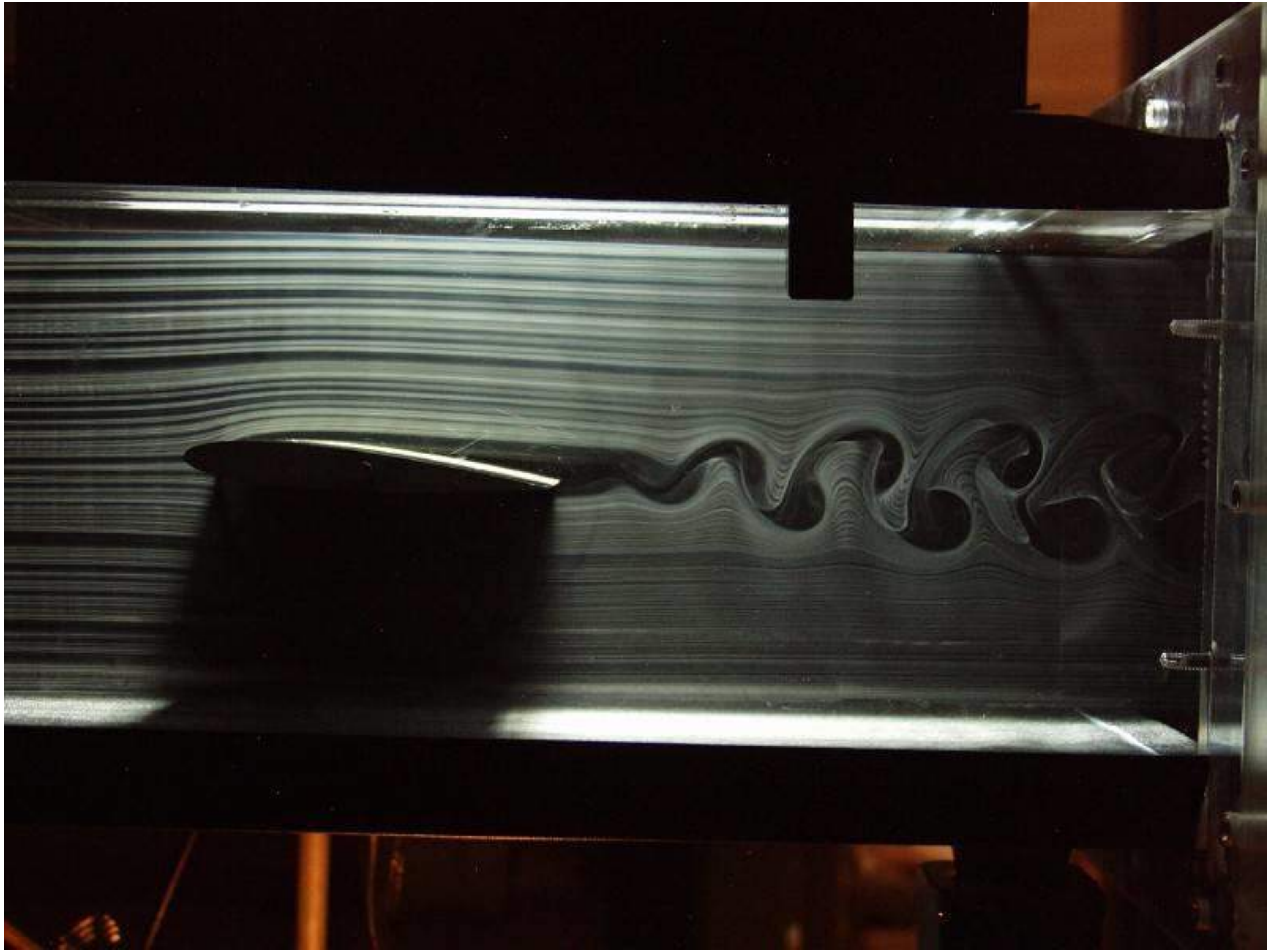
Stall

- In fluid dynamics, a stall is a reduction in the lift coefficient generated by a foil as angle of attack increases.
- Stalls in fixed-wing flight are often experienced as a sudden reduction in lift as the pilot increases the wing's angle of attack and exceeds its critical angle of attack
- A stall does not mean that the engine(s) have stopped working, or that the aircraft has stopped moving — the effect is the same even in an unpowered glider aircraft.

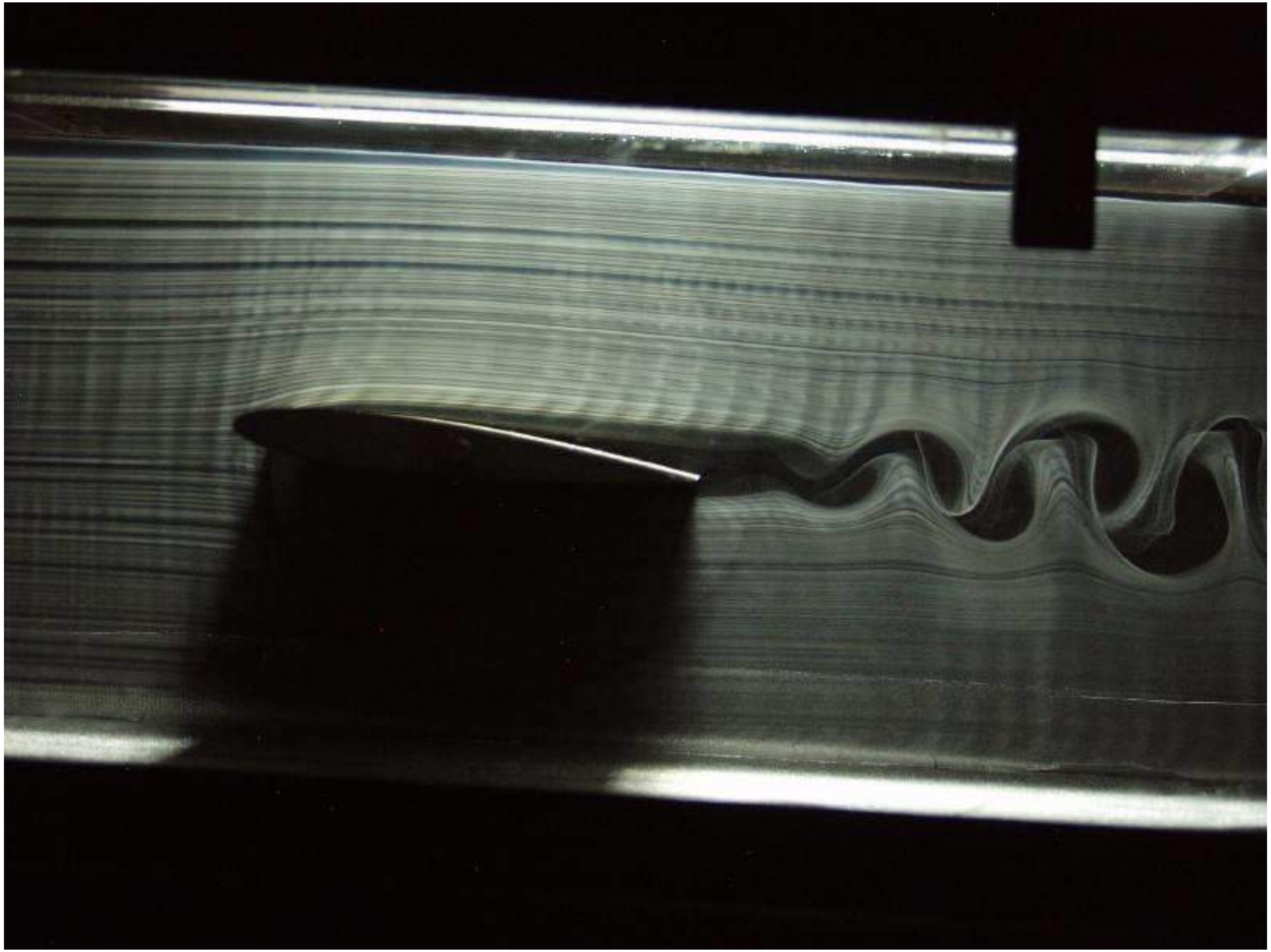
- A stall is a condition in aerodynamics and aviation wherein the angle of attack increases beyond a certain point such that lift begins to decrease. The angle at which this occurs is called the *critical angle of attack*.
- This critical angle is dependent upon the airfoil section or profile of the wing, its planform, its aspect ratio, and other factors.
- Flow separation begins to occur at small angles of attack while flow over the wing is still dominant. As angle of attack increases, the separated regions on the top of the wing increase in size and hinder the wing's ability to create lift. At the critical angle of attack, separated flow is so dominant that additional increases in angle of attack produce less lift .

Airflow separating from a wing at a high angle of attack

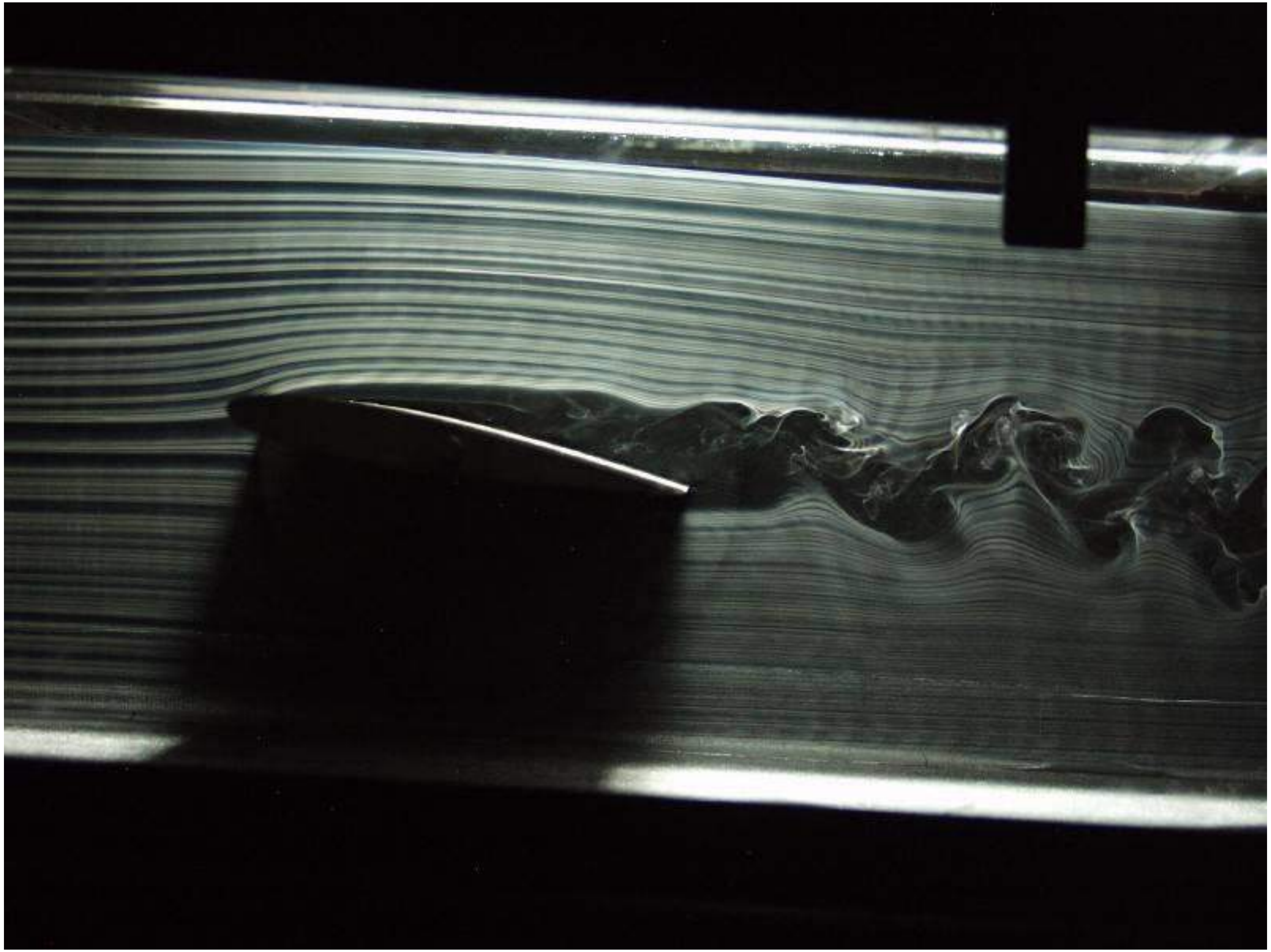
AOA = 2°



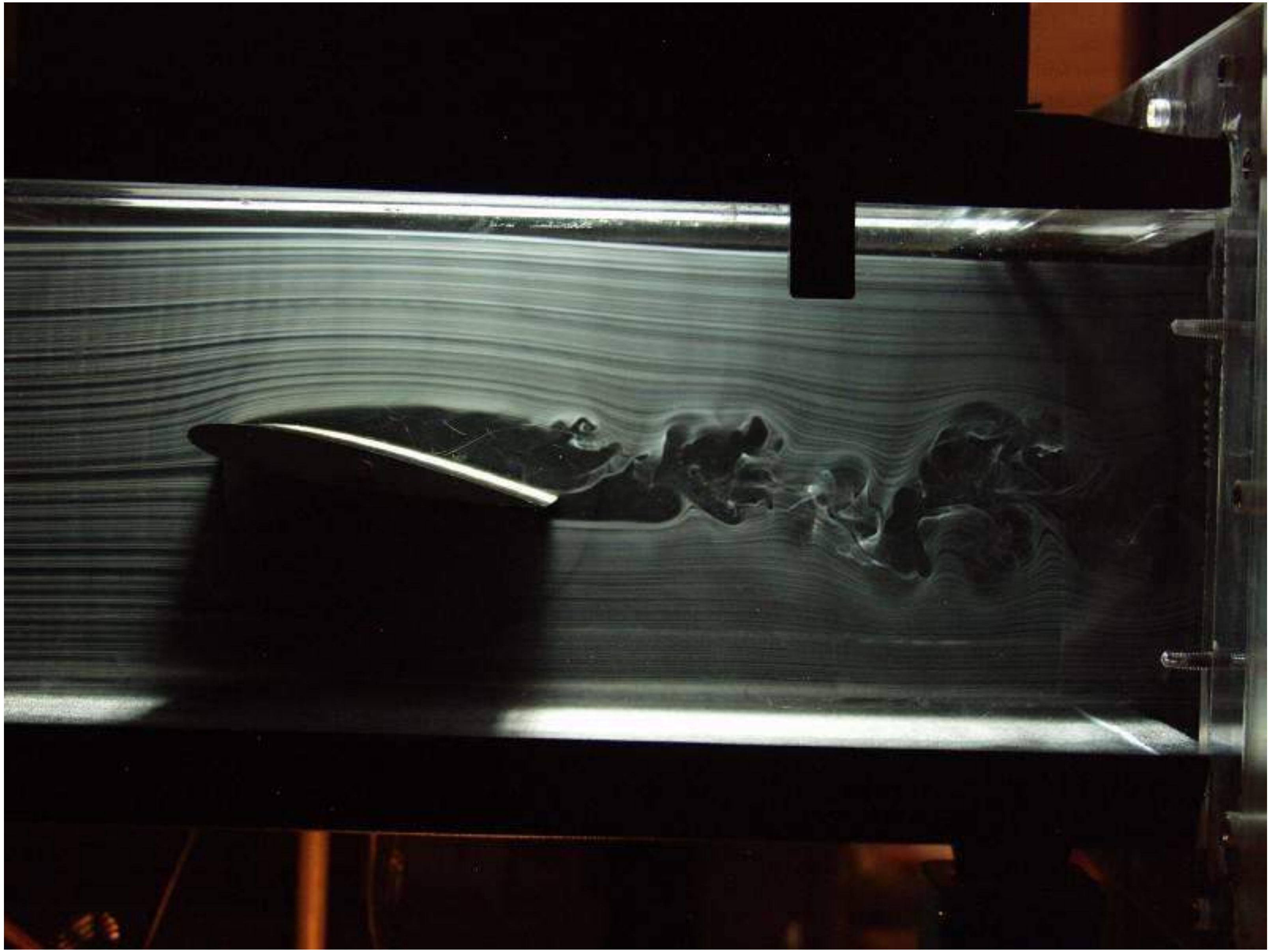
AOA = 3°



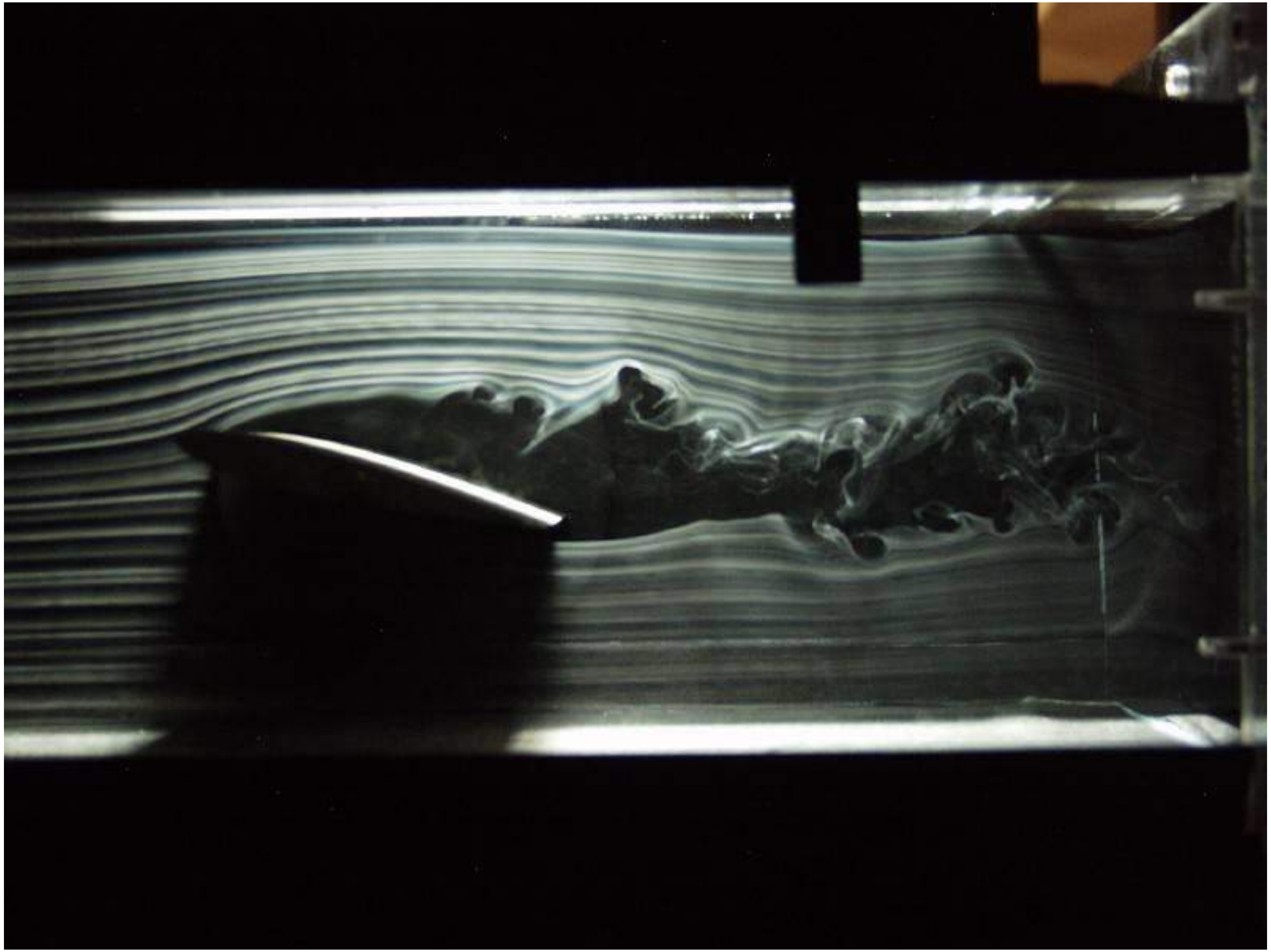
AOA = 6°



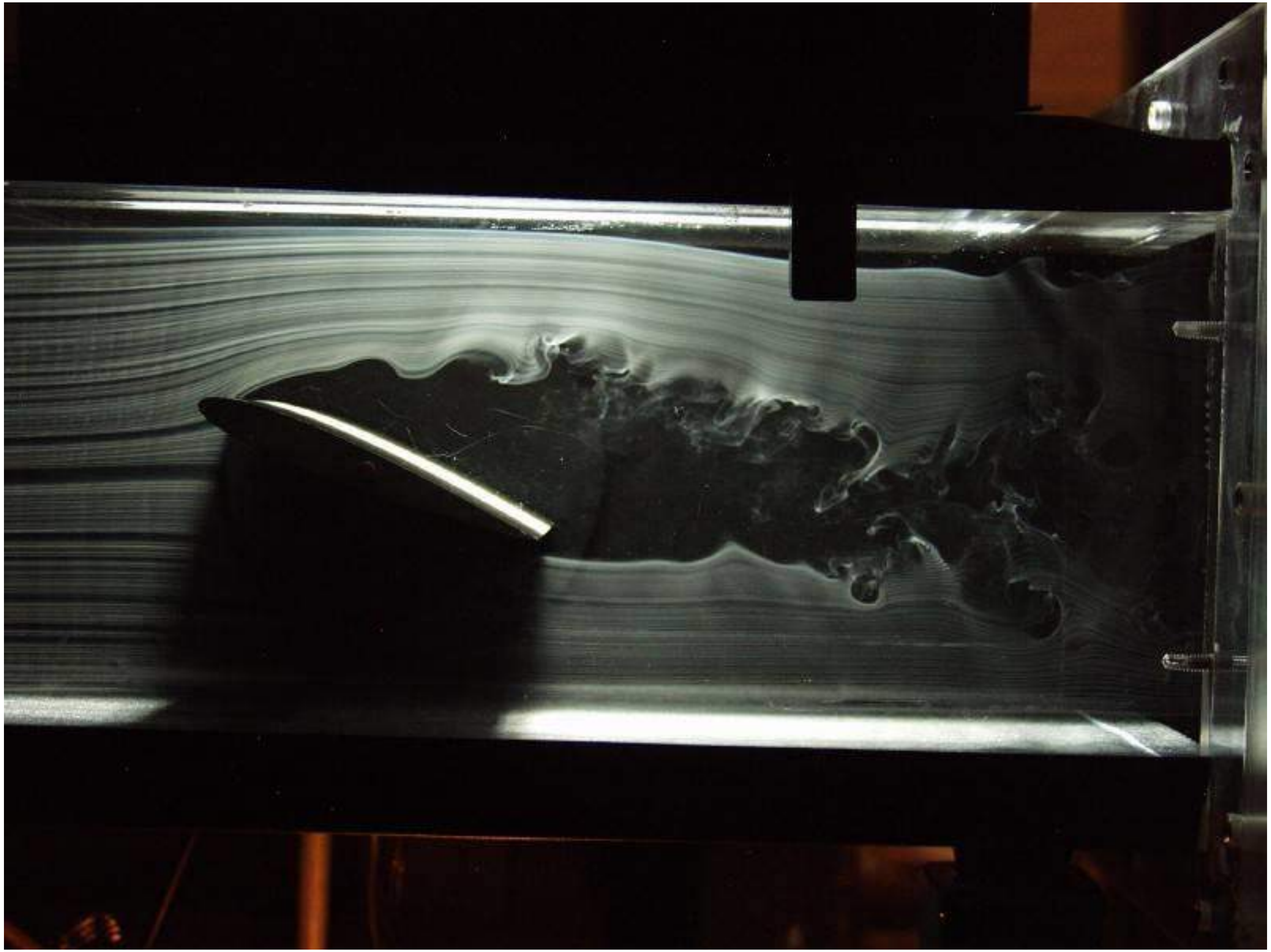
AOA = 9°



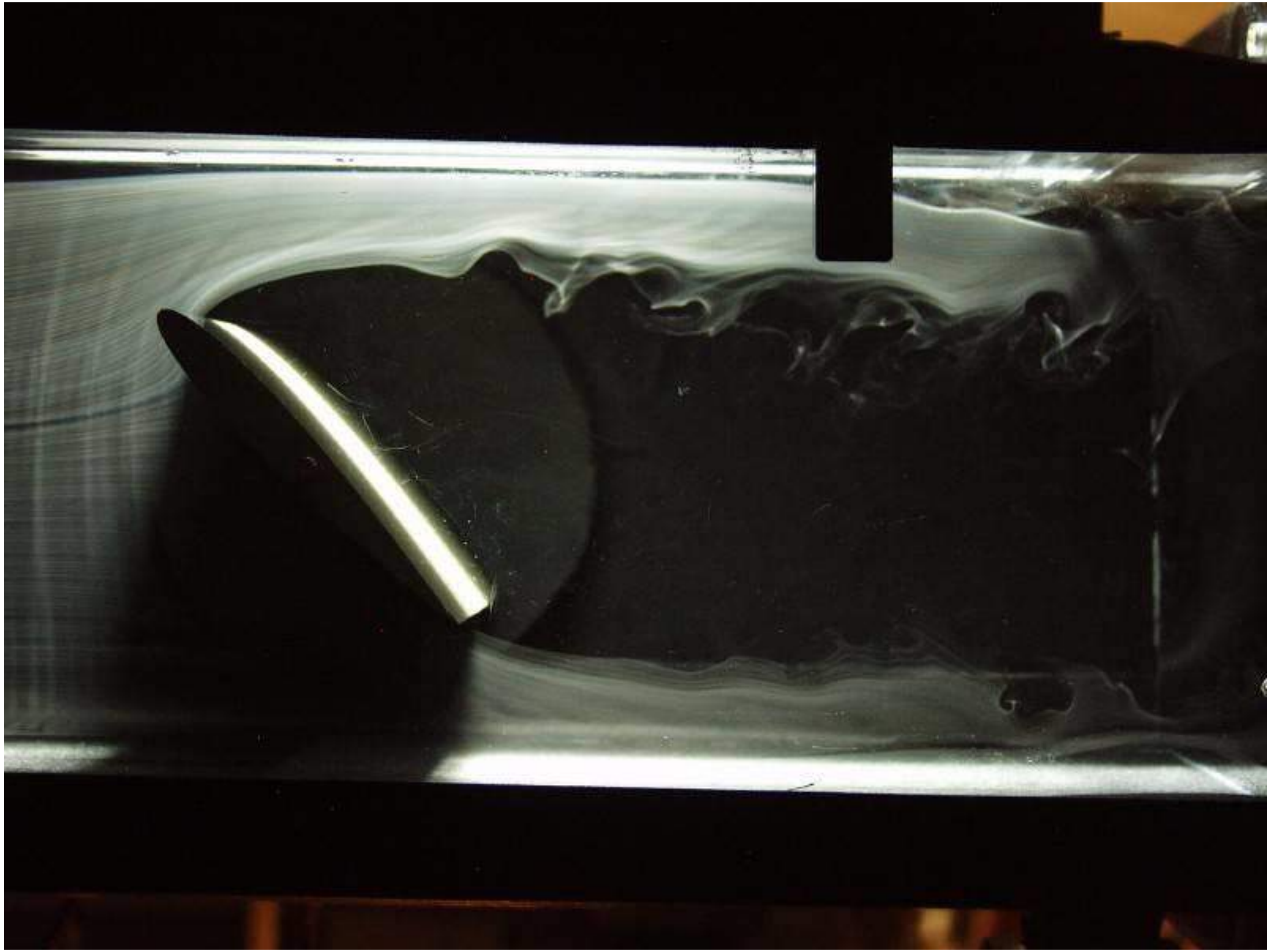
AOA = 12°



AOA = 20°



AOA = 60°



- As the AOA increases beyond α_{max}, separation of the airflow from the upper surface of the wing becomes more significant, causing the reduction of C_L.
- At the critical AOA, the wing is unable to support the weight of the aircraft, causing the aircraft to descend, which in turn, causes the AOA to increase further. This is known as STALL.

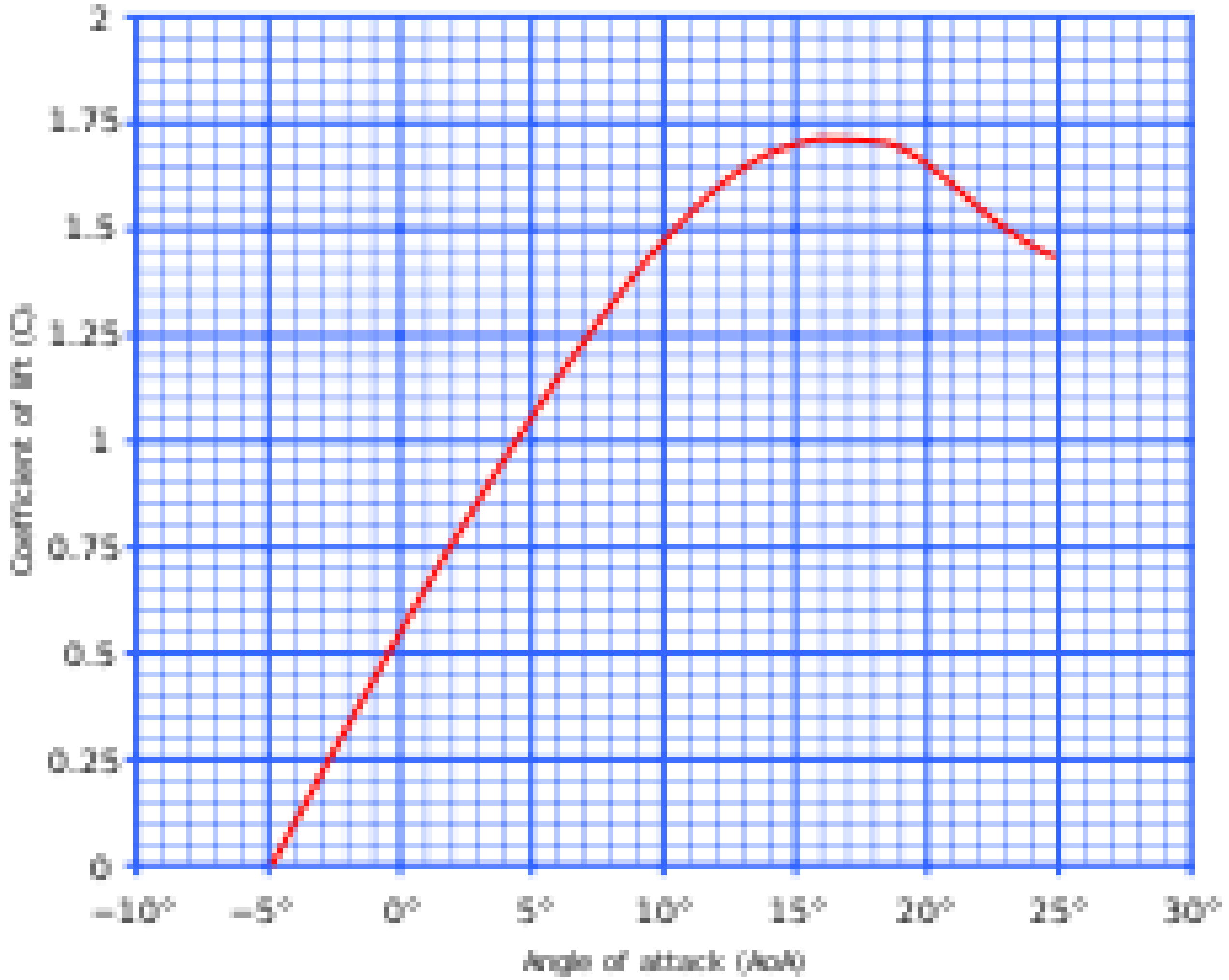
- The airspeed at which the aircraft stalls depends on many factors like—weight of the aircraft, the load factor at the time and the thrust from engine.
- The critical AOA is typically at 15° for many aerofoils.
- Most of the modern aircraft have a stall warning system.
- The stall warning system gives warning about the incipient stall condition, by alerting the pilot even before stall conditions are reached.

High AOA

- Very high angles of attack in fighter aircraft give an unsurpassing agility to evade missile hits.
- Indian–Russian fighter aircraft SU-30MKI has the highest AOA of 123° for a duration of 3 seconds. The manoeuvre is called Pugachev's (after the Russian pilot) Cobra manoeuvre.

- Using additional control surfaces, vectored thrust by engines, and the high-lift devices, like leading extensions, flyable AOA has been substantially increased from about 20° to over 45°, and in some aircraft AOA can become 90º, i.e. wing is perpendicular to the direction of motion.

- Any yaw of the aircraft as it enters the stall regime can result in autorotation, which is also sometimes referred to as a "spin".
- As air no longer flows smoothly over the wings during a stall, aileron control of roll becomes less effective, while simultaneously the tendency for the ailerons to generate adverse yaw increases.
- This characteristic increases the likelihood of the aircraft entering into a spin

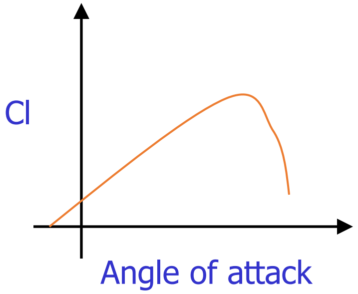


An example of the relationship between angle of attack and lift on a cambered airfoil. The exact relationship is usually measured in a wind tunnel and depends on the airfoil section. The relationship for an aircraft wing depends on its aspect ratio.

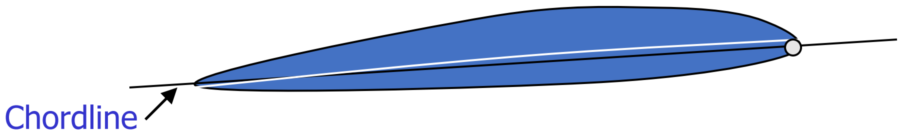
- The graph shows that the greatest amount of lift is produced as the critical angle of attack is reached
- This angle is 17.5 degrees in this case but changes from airfoil to airfoil.
- In particular, for aerodynamically thick airfoils the critical angle is higher than with a thin airfoil of the same camber.
- Symmetric airfoils have lower critical angles .
- The graph shows that, as the angle of attack exceeds the critical angle, the lift produced by the airfoil decreases.

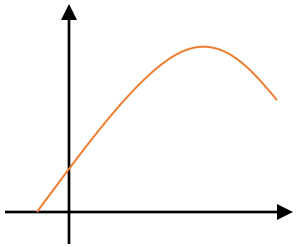
- The slower an airplane goes, the more angle of attack it needs to produce lift equal to the aircraft's weight.
- As the speed decreases further, at some point this angle will be equal to the critical (stall) angle of attack. This speed is called the "stall speed".

Basics on airfoils and lift generation

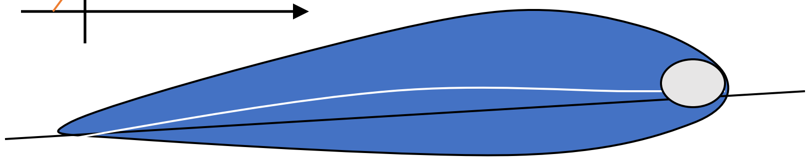


Highspeed airfoil that will generate low Cl and low Cd. Rather sharp stall at low angle of attack





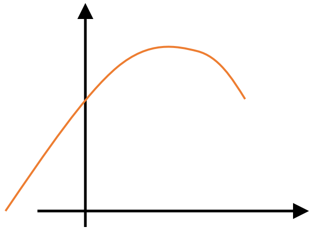
Airfoil for medium speeds that will produce more lift, but also more drag
"Nice " stall

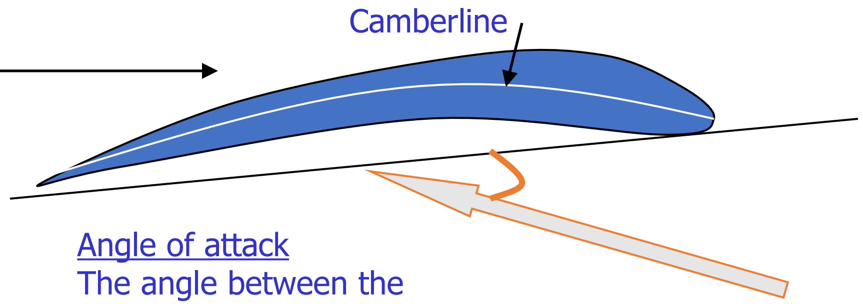


Smooth stall characteristics depends largely on :

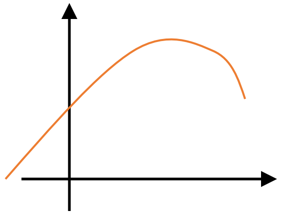
- Large nose radius
- High camber
- Medium to high profile thickness

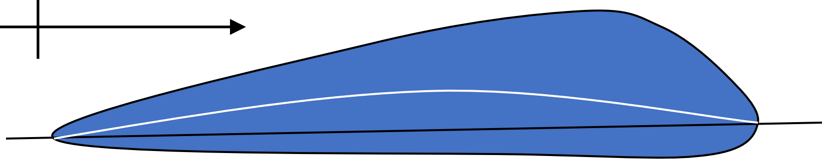
Low speed airfoil, will produce much lift at low angles of attack, much drag and smooth stall at medium angle of attack





Angle of attack
The angle between the chordline and flightpath





"Paraglider airfoil"
A lot of volume gives much lift and smooth stalling at high angle of attack
Also much drag as airspeed increases.

The lift equation

$$Lift = C_L \tfrac{1}{2} \rho v^2 A$$

The values in the expression are:

- ρ (the Greek letter rho) is the density of the air, in kg/m³
- **v^2** is the flight speed in meters per second
- **A** is the wing area in square meters
- **C_L** is a dimensionless quantity – the lift coefficient. Mostly depends on the ANGLE of attack and the SHAPE of the wing.

The Drag Equation

$$Drag = C_D \, {}^{1}\!/_{2} \, \rho v^2 A$$

The drag equation is similar to the lift equation with the exception that we have a DRAG COEFFICIENT rather than a LIFT COEFFICIENT. As C_L depending on "aoa", the C_D depends on the SQUARE of the "aoa".

INDUCED DRAG: Newtonian & Pressure Induced

Induced drag is the unavoidable by-product of lift and increases as the angle of attack increases

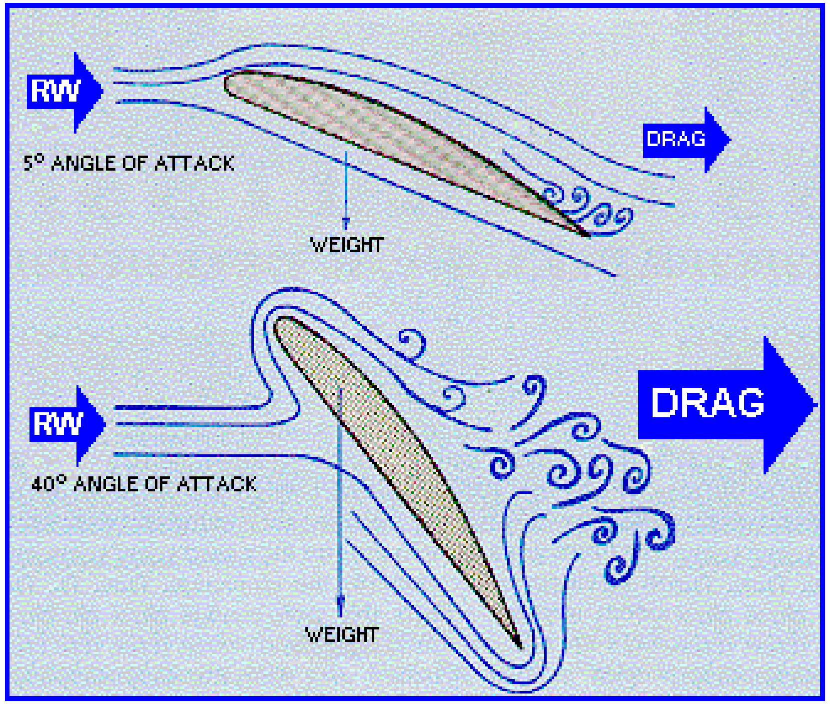


Figure 4-8 Dynamic drag.

Newtonian or DYNAMIC DRAG is caused by the INERTIA of AIR.

Pressure Induced Drag occurs when the “aoa” is too large and the air Flow becomes turbulent.

PARASITE DRAG

All drag other than induced drag is parasite drag.

Skin-friction drag is caused by the friction between outer surfaces of the aircraft and the air through which it moves. It will be found on all surfaces of the aircraft: wing, tail, engine, landing gear, and fuselage

Lift / Drag

The LIFT/DRAG ratio can be found by taking the lift coefficient and dividing by the drag coefficient.

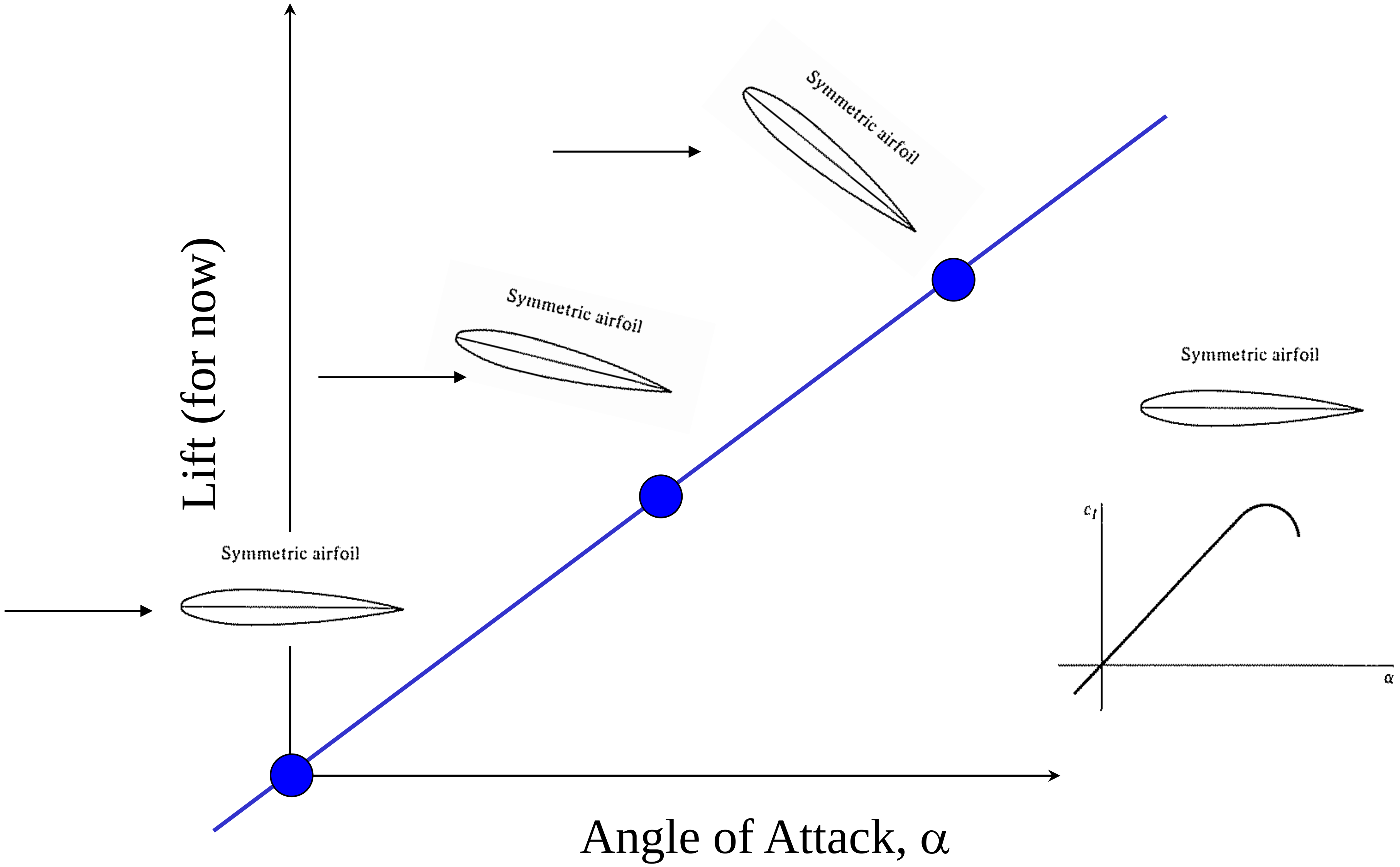
$$L/D\ ratio = \frac{C_L}{C_D}$$

The L/D ratio is a measure of EFFICIENCY!!!

L/D Ratio

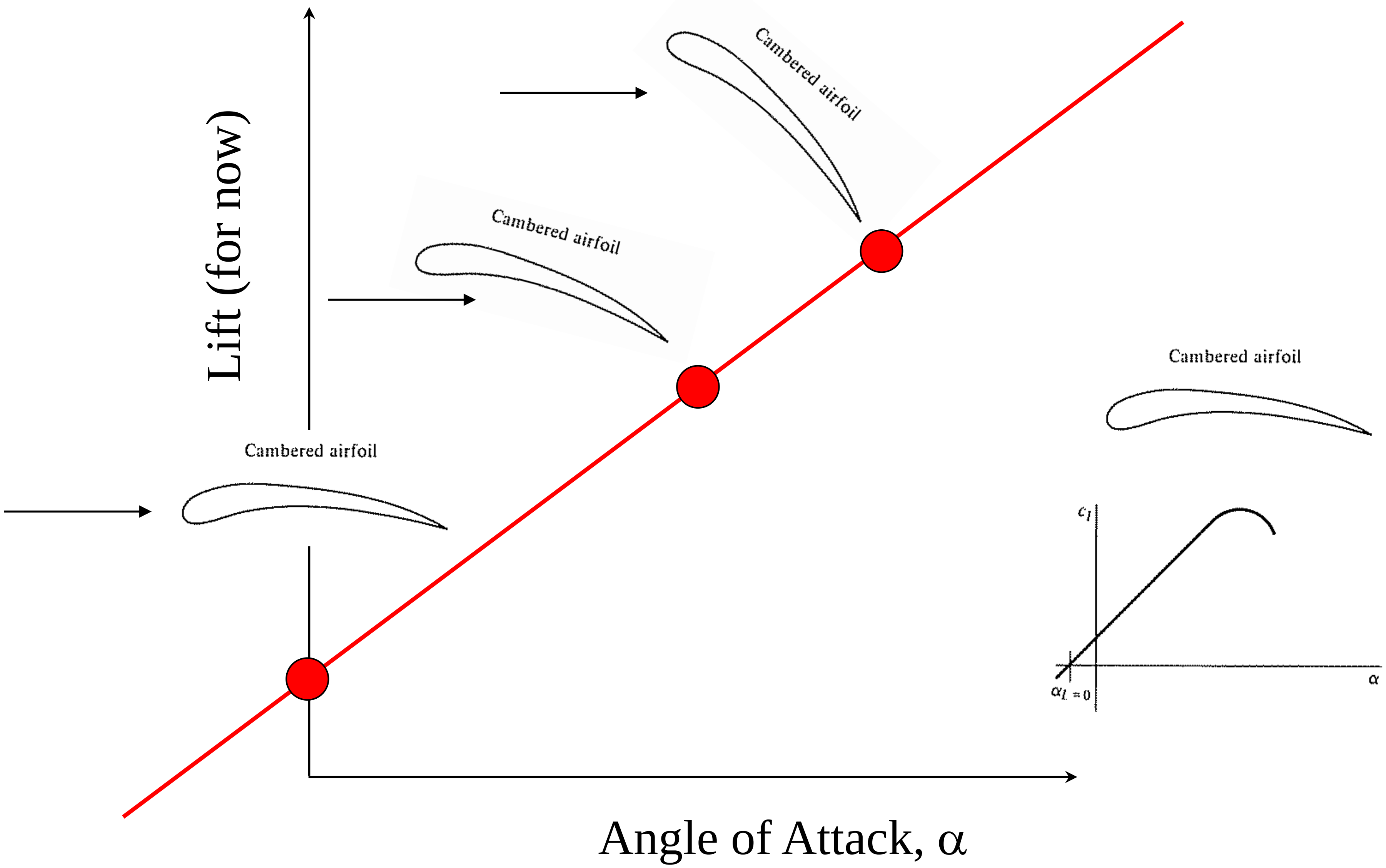
Because lift and drag are both aerodynamic forces, we can think of the L/D ratio as an aerodynamic efficiency factor for the aircraft. Designers of gliders and designers of cruising aircraft want a high L/D ratio to maximize the distance which an aircraft can fly.

SAMPLE DATA: SYMMETRIC AIRFOIL



A symmetric airfoil generates zero lift at zero α

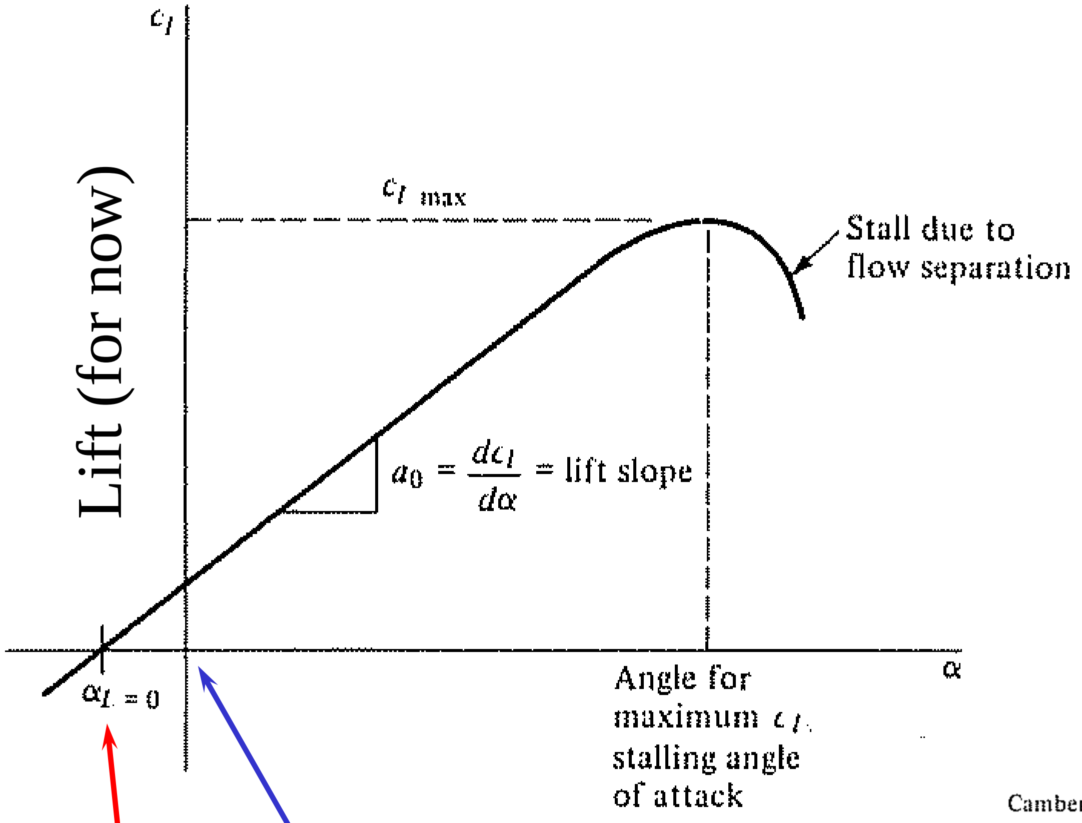
SAMPLE DATA: CAMBERED AIRFOIL



A cambered airfoil generates positive lift at zero α

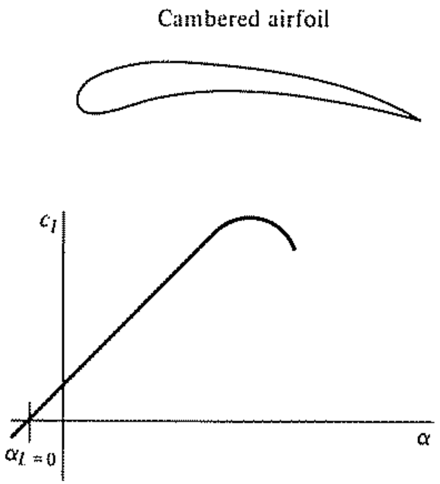
SAMPLE DATA

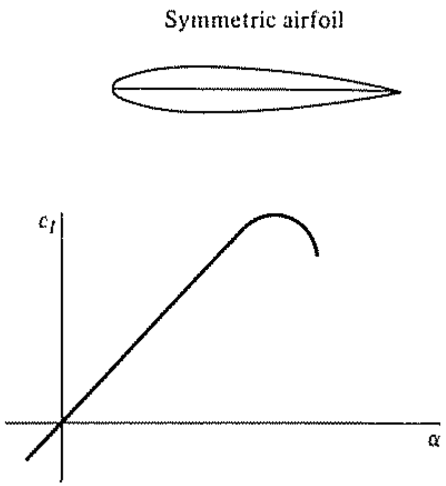
- Lift coefficient (or lift) linear variation with angle of attack, a
  - Cambered airfoils have positive lift when α = 0
  - Symmetric airfoils have zero lift when α = 0
- At high enough angle of attack, the performance of the airfoil rapidly degrades → stall



Cambered airfoil has lift at α=0

At negative α airfoil will have zero lift





TYPES OF FLOWS: FRICTION VS. NO-FRICTION

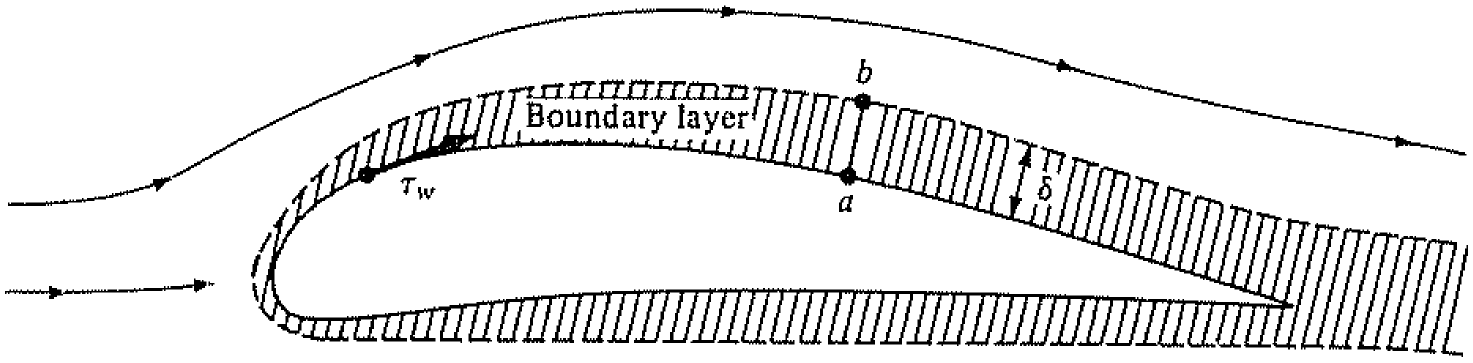
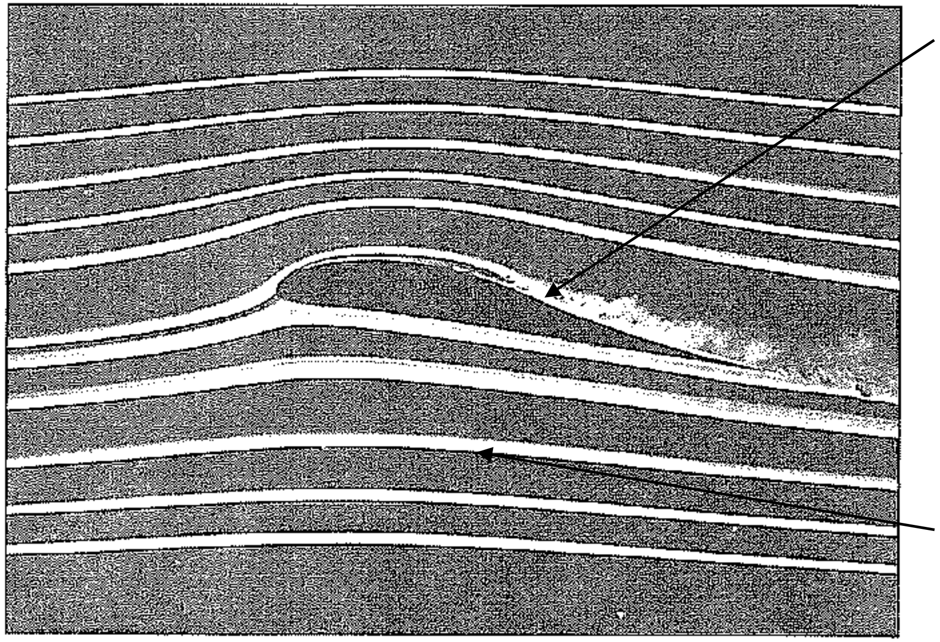


Figure 4.32 Flow in real life, with friction. The thickness of the boundary layer is greatly overemphasized for clarity



Flow very close to surface of airfoil is Influenced by friction and is viscous (boundary layer flow)

Stall (separation) is a viscous phenomena

Flow away from airfoil is not influenced by friction and is wholly inviscid

THE REYNOLDS NUMBER, Re

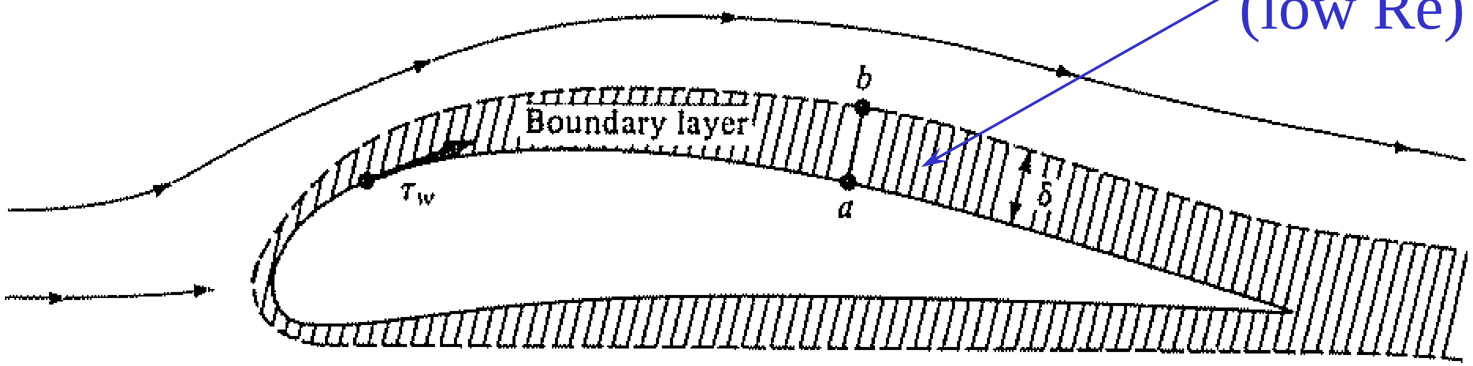
- One of most important dimensionless numbers in fluid mechanics/ aerodynamics
- Reynolds number is ratio of two forces:
  - Inertial Forces
  - Viscous Forces
  - c is length scale (chord)

$$\mathrm{Re} = \frac{\rho V_\infty c}{\mu}$$

- Reynolds number tells you when viscous forces are important and when viscosity may be neglected

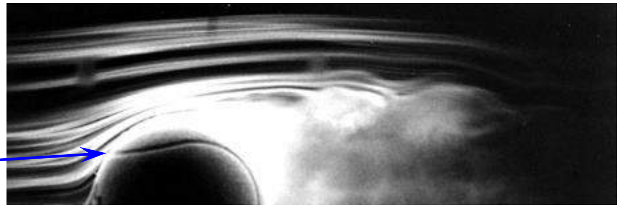
Outside B.L. flow
Inviscid (high Re)

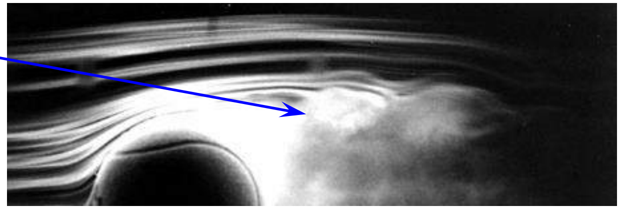
Within B.L. flow
highly viscous
(low Re)



SUMMARY OF VISCOUS EFFECTS ON DRAG

- Friction has two effects:
  1. Skin friction due to shear stress at wall
  2. Pressure drag due to flow separation





$$D = D_{friction} + D_{pressure}$$

Total drag due to viscous effects = Drag due to skin friction + Drag due to separation

Called **Profile Drag**

Drag due to skin friction	Drag due to separation
Less for laminar More for turbulent	More for laminar Less for turbulent

TRUCK SPOILER EXAMPLE

- Note ‘messy’ or turbulent flow pattern
- High drag
- Lower fuel efficiency

- Spoiler angle increased by + 5°
- Flow behavior more closely resembles a laminar flow
- Tremendous savings (< $10,000/yr) on Miami-NYC route

LIFT, DRAG, AND MOMENT COEFFICIENTS

- Behavior of L, D, and M depend on α, but also on velocity and altitude
  - V_∞, ρ_∞, Wing Area (S), Wing Shape, μ_∞, compressibility
- Characterize behavior of L, D, M with coefficients (c_l, c_d, c_m)

$$L = \frac{1}{2}\rho V_\infty^2 S c_l$$

$$c_l \equiv \frac{L}{\frac{1}{2}\rho V_\infty^2 S} = \frac{L}{q_\infty S}$$

$$c_l = f\left(\alpha, M_\infty, \mathrm{Re}\right)$$

LIFT, DRAG, AND MOMENT COEFFICIENTS

- Behavior of L, D, and M depend on α, but also on velocity and altitude
  - V_∞, ρ_∞, Wing Area (S), Wing Shape, μ_∞, compressibility

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$$\boxed{L = \frac{1}{2}\rho V_\infty^2 S c_l}$$

$$c_l \equiv \frac{L}{\frac{1}{2}\rho V_\infty^2 S} = \frac{L}{q_\infty S}$$

$$c_l = f_1(\alpha, M_\infty, \mathrm{Re})$$

$$\boxed{D = \frac{1}{2}\rho V_\infty^2 S c_d}$$

$$c_d \equiv \frac{D}{\frac{1}{2}\rho V_\infty^2 S} = \frac{D}{q_\infty S}$$

$$c_d = f_2(\alpha, M_\infty, \mathrm{Re})$$

$$\boxed{M = \frac{1}{2}\rho V_\infty^2 S c c_m}$$

$$c_m \equiv \frac{M}{\frac{1}{2}\rho V_\infty^2 S c} = \frac{L}{q_\infty S c}$$

$$c_m = f_3(\alpha, M_\infty, \mathrm{Re})$$

We use lower case, c_l, c_d, and c_m for **infinite** wings (airfoils)
We use upper case, C_L, C_D, and C_M for **finite** wings

PRESSURE DISTRIBUTION AND LIFT

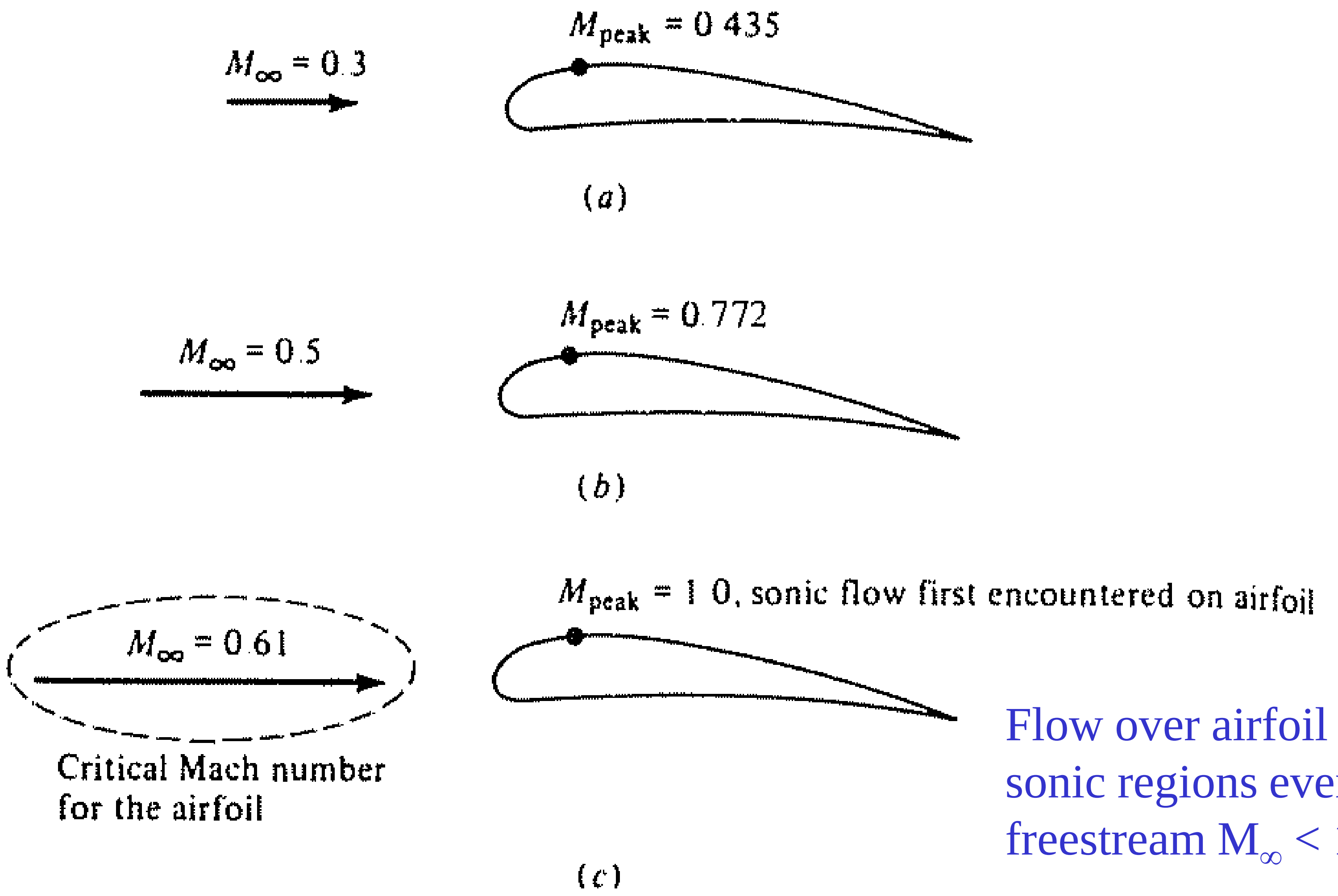
- Lift comes from pressure distribution over top (suction surface) and bottom (pressure surface)

- Lift coefficient also result of pressure distribution

CRITICAL MACH NUMBER AND CRITICAL PRESSURE COEFFICIENT

- As the gas expands around the top surface near the leading edge, the velocity and hence the Mach number will increase rapidly.
- The flow over an airfoil can locally be sonic (or higher) even though the freestream Mach number is subsonic.
- The free-stream Mach number at which sonic flow is first obtained somewhere on the airfoil surface is called the *critical Mach number Mcr* of the airfoil.

- As air expands around top surface near leading edge, velocity and M will increase
- Local M > M_∞



Flow over airfoil may have sonic regions even though freestream $M_\infty < 1$

- The point on the airfoil where the local M is a peak value is also the point of minimum surface pressure.
- The pressure coefficient, C_P will correspondingly have its most negative value at this point.
- Thinner airfoil has higher critical Mach number

WAVE DRAG (AT SUPERSONIC SPEEDS)

- When Mα is supersonic, a major new physical phenomenon is introduced: shock waves.
- With respect to airfoils (as well as all other aerodynamic bodies), shock waves in supersonic flow create a new source of drag, called *wave drag.*

- Imagine that we have a small source of sound waves: a tiny "beeper" (something like a tuning fork). At time $t = 0$ assume that the beeper is at point P in Fig.
- At this point let the beeper emit a sound wave, which will propagate in all directions at the speed of sound *a*. Also let the beeper move with velocity *V*, where V is less than the speed of sound. At time *t*, the sound wave will have moved outward by a distance *at*, as shown in Fig.
- At the same *time t*, the beeper will have moved a distance *Vt* to point *Q*. Because $V < a$, the beeper will always stay inside the sound wave.
- If the beeper is constantly emitting sound waves as it moves along, these waves will constantly move outward, ahead of the beeper. As long as $V < a$, the beeper will always be inside the envelope formed by the sound waves.

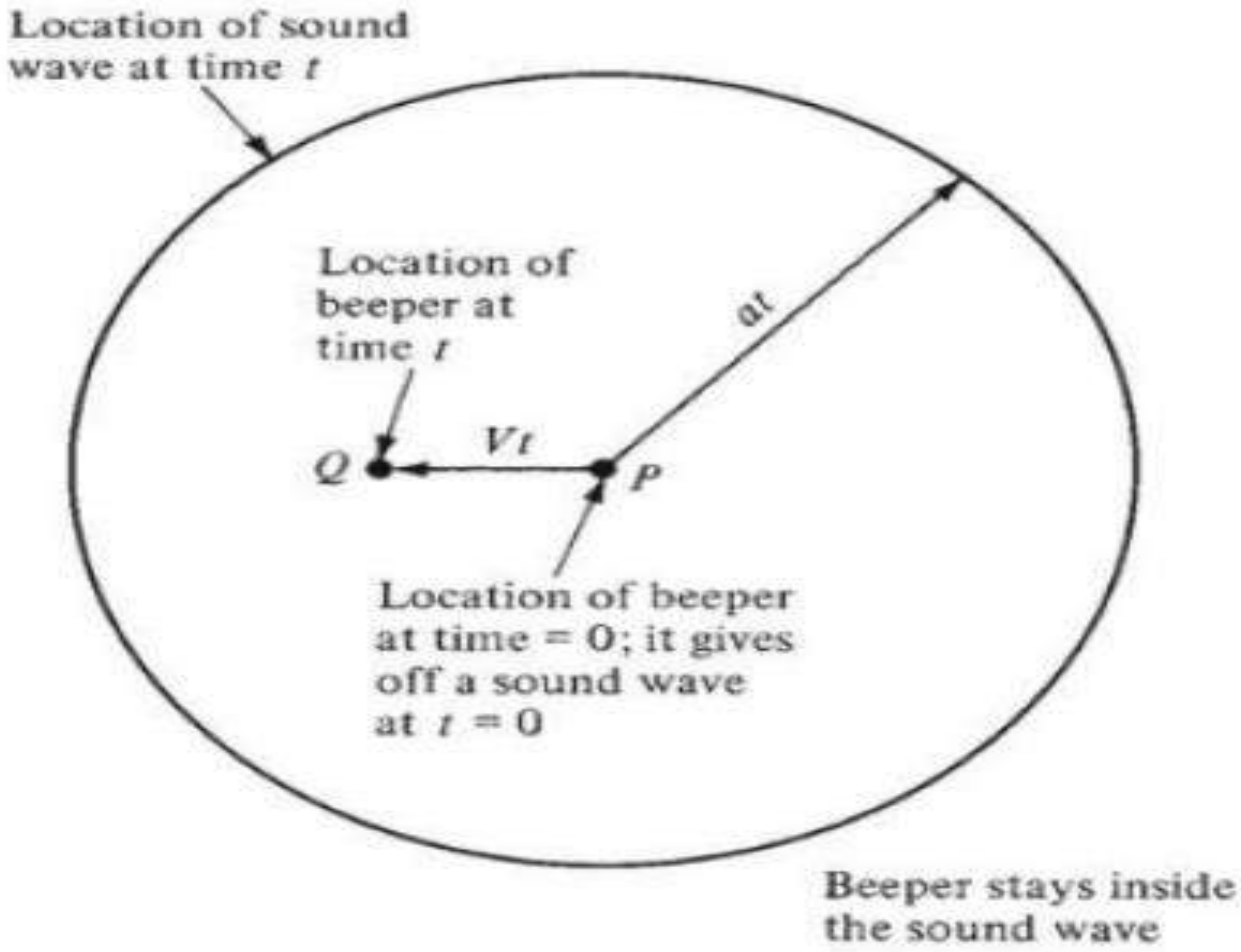


Figure 5.34 Beeper moving at less than the speed of sound.

- Now, we change the situation: assume that the beeper is moving at supersonic speed; that is, $V > a$.
- At time,t= 0, assume that the beeper is at point R in Fig.
- At this point let the beeper emit a sound wave, which, as before, will propagate in all directions at the speed of sound a.
- At time t, the sound wave will have moved outward by a distance at, as shown in Fig..
- At the same time t, the beeper will have moved a distance Vt to point S.
- However, because $V > a$, the beeper will now be outside the sound wave.

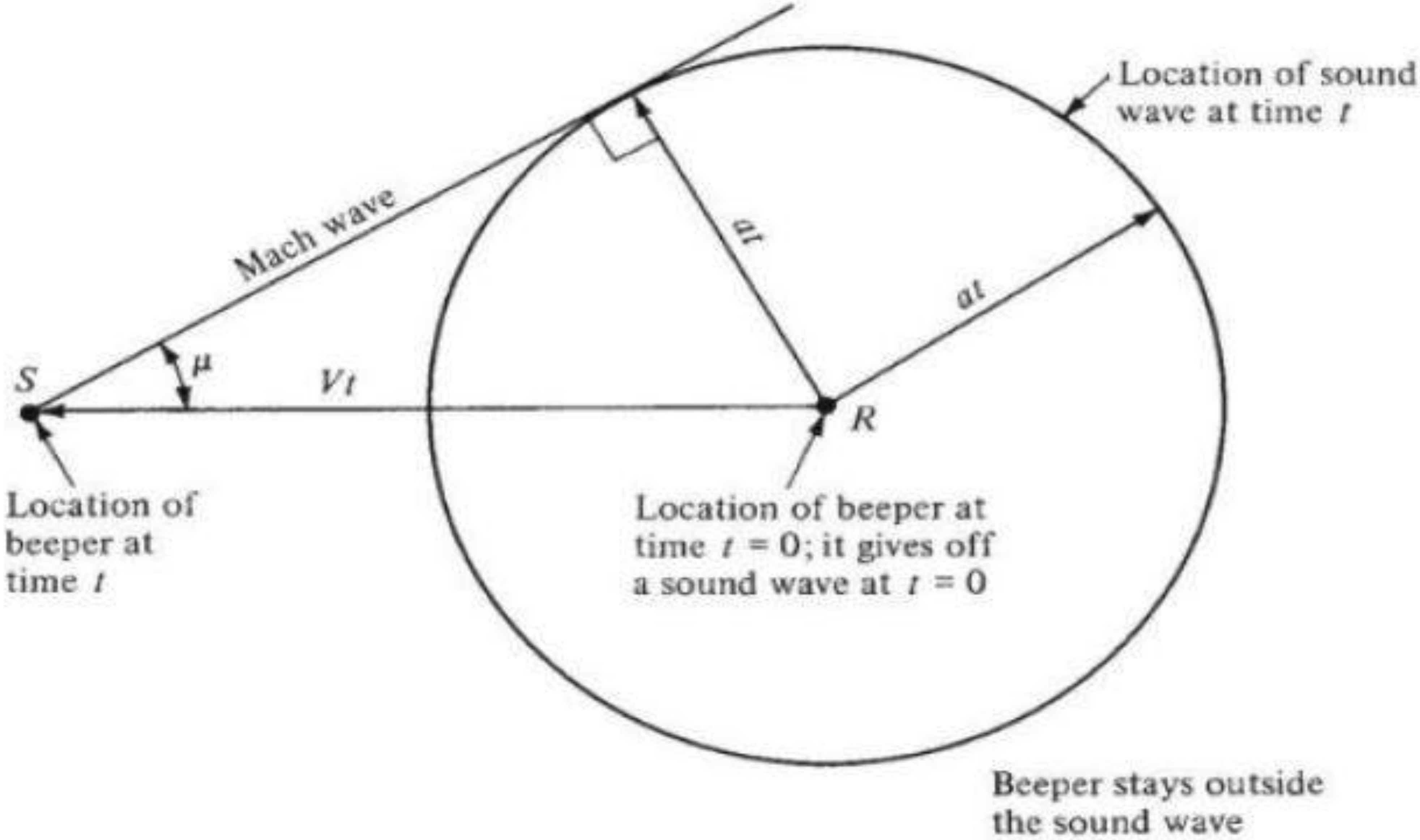


Figure 5.35 The origin of Mach waves and shock waves. The beeper is moving faster than the speed of sound.

- If the beeper is constantly emitting sound waves as it moves along, these waves will now pile up inside an envelope formed by a line from point *S* tangent to the circle formed by the first sound wave, centered at point *R*.
- This tangent line, the line where the pressure disturbances are piling up, is called a *Mach wave.*
- The vertex of the wave is fixed to the moving beeper at point *S*.

- In supersonic flight, the air ahead of the beeper has no warning of the approach of the beeper.
- Only the air behind the Mach wave has felt the presence of the beeper, and this presence is communicated by pressure (sound) waves confined inside the conical region bounded by the Mach wave.
- In contrast, in subsonic flight, the air ahead of the beeper is forewarned about the oncoming beeper by the sound waves. In this case there is no piling up of pressure waves; there is no Mach wave.

- In Fig. the Mach wave that is formed makes an angle with the direction of movement of the beeper.
- This angle, defined as the *Mach angle,* is easily obtained from the geometry of Fig.

$$\sin\mu = \frac{at}{Vt} = \frac{a}{V} = \frac{1}{M}$$

$$\boxed{\text{Mach angle} \equiv \mu \equiv \arcsin\frac{1}{M}}$$

- In real life, a very thin object (such as a thin needle) moving at Mα > 1 creates a very weak disturbance in the flow, limited to a Mach wave.

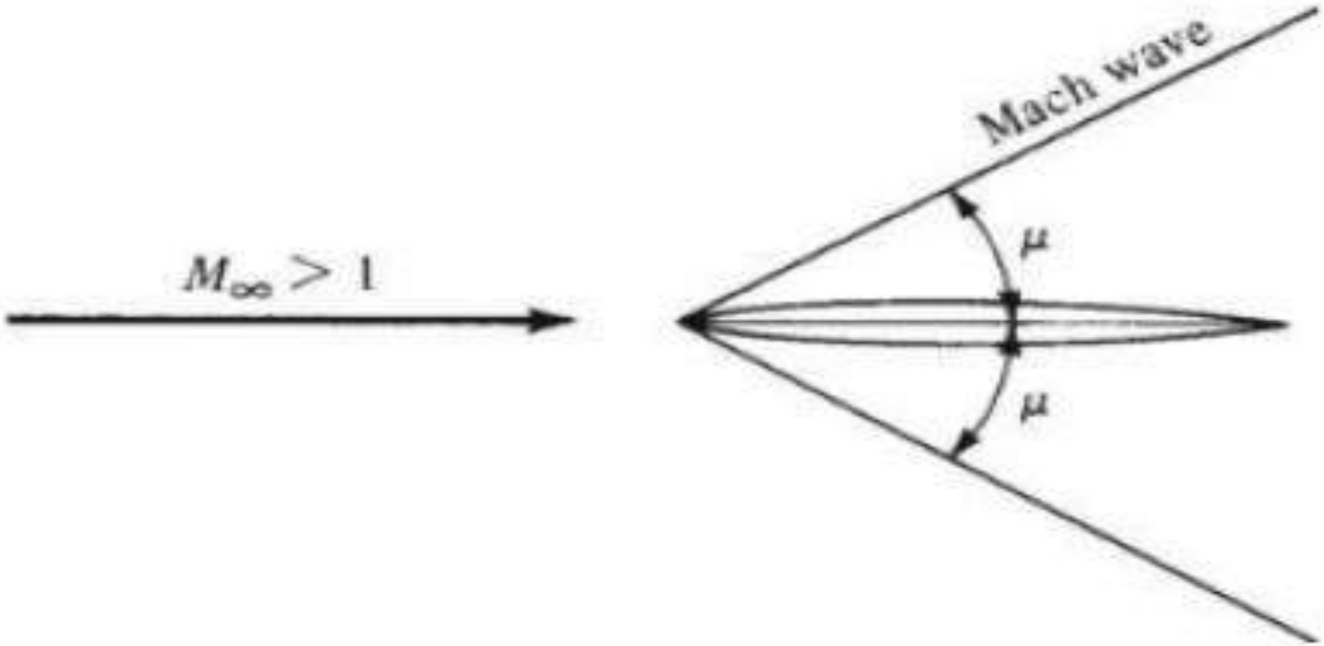


Figure 5.36 Mach waves on a needlelike body.

- A thicker object such as the wedge moving at supersonic speeds will create a strong disturbance, called a *shock wave*.
- The shock wave will be inclined at an oblique angle *beta*, where beta > μ..

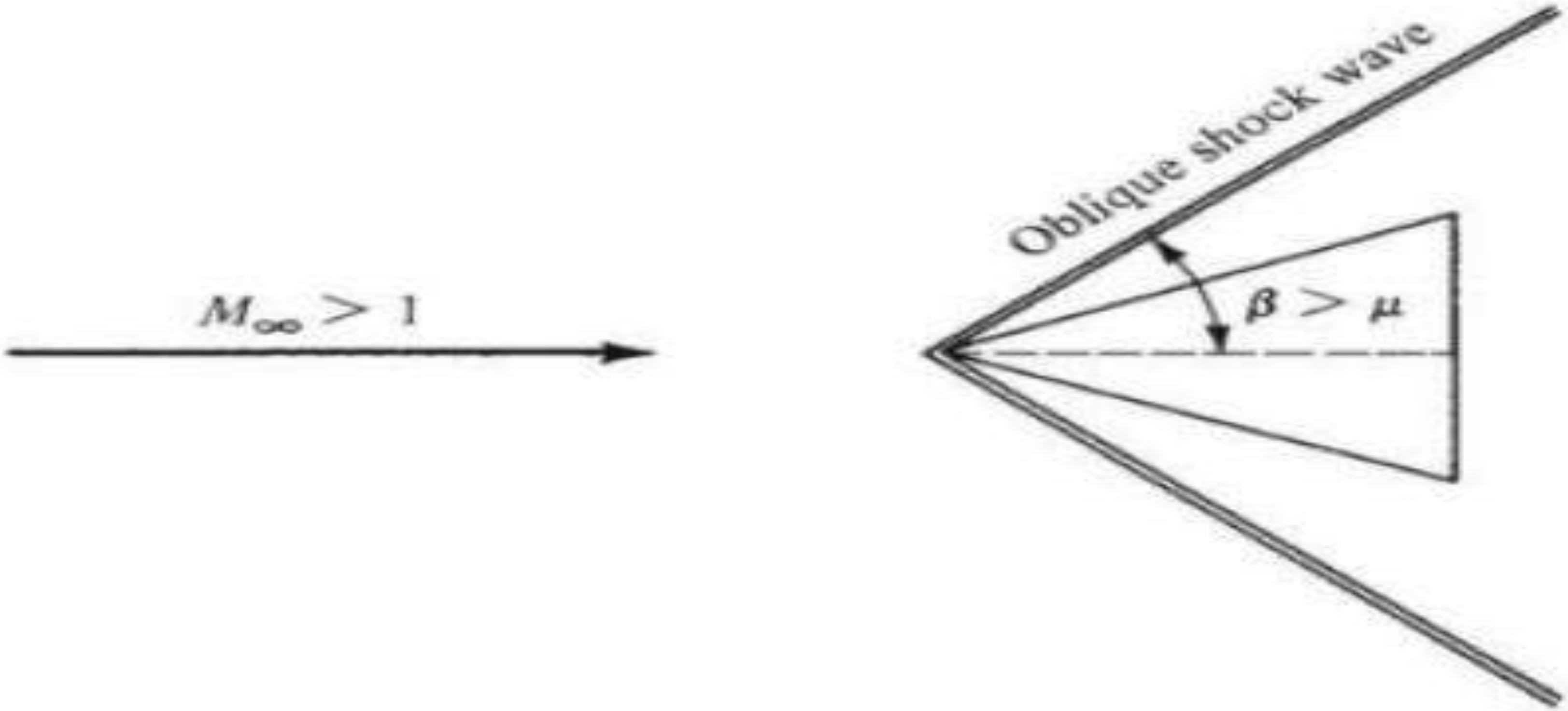


Figure 5.37 Oblique shock waves on a wedge-type body.

- Consider now the pressure on the surface of the wedge, as sketched in Fig.
- Because *p* increases across the oblique shock wave, at the wedge surface, p > Pα ·
- Because the pressure acts normal to the surface and the surface itself is inclined to the relative wind, a net drag will be produced on the wedge.
- This drag is called *wave drag* because it is inherently due to the pressure increase across the shock wave.
- To minimize the strength of the shock wave, all supersonic airfoil profiles are thin, with relatively sharp leading edges.

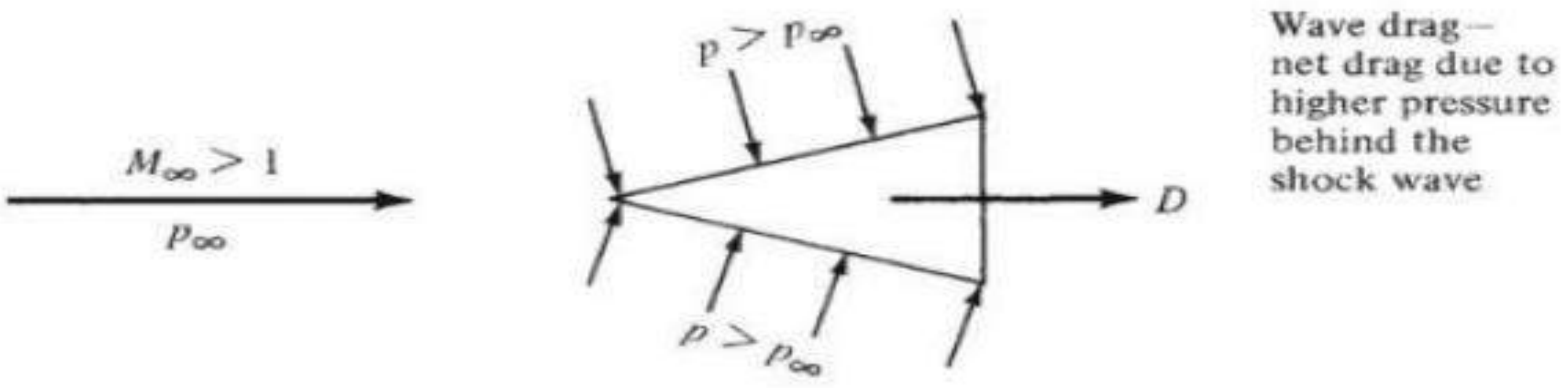


Figure 5.38 Pressure distribution on a wedge at supersonic speeds; origin of wave drag.

SUMMARY OF AIRFOIL DRAG

$$D = D_f + D_p + D_w$$

where

D = total drag on airfoil

D_f = skin friction drag

D_p = pressure drag due to flow separation

D_w = wave drag (present only at transonic and supersonic speeds; zero for subsonic speeds below the drag-divergence Mach number)

In terms of the drag coefficients, we can write

$$c_d = c_{d,f} + c_{d,p} + c_{d,w}$$

where c_d, $c_{d,f}$, $c_{d,p}$, and $c_{d,w}$ are the total drag, skin friction drag, pressure drag, and wave drag coefficients, respectively. The sum $c_{d,f} + c_{d,p}$ is called the *profile drag coefficient;* this is the quantity given by the data in App. D. The profile drag coefficient is relatively constant with M_∞ at subsonic speeds.

- Approximate relations for the lift and wave drag coefficients are,

$$c_l = \frac{4\alpha}{(M_\infty^2 - 1)^{1/2}} \tag{5.50}$$

$$c_{d,w} = \frac{4\alpha^2}{(M_\infty^2 - 1)^{1/2}} \tag{5.51}$$

α is in radians

FINITE WINGS

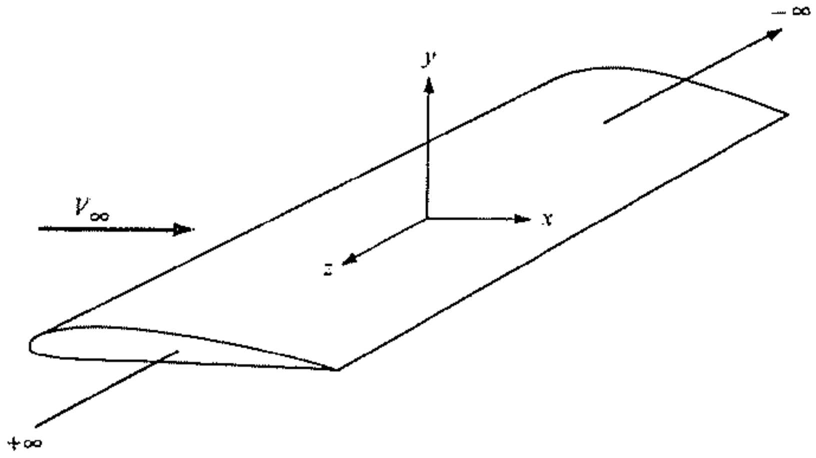
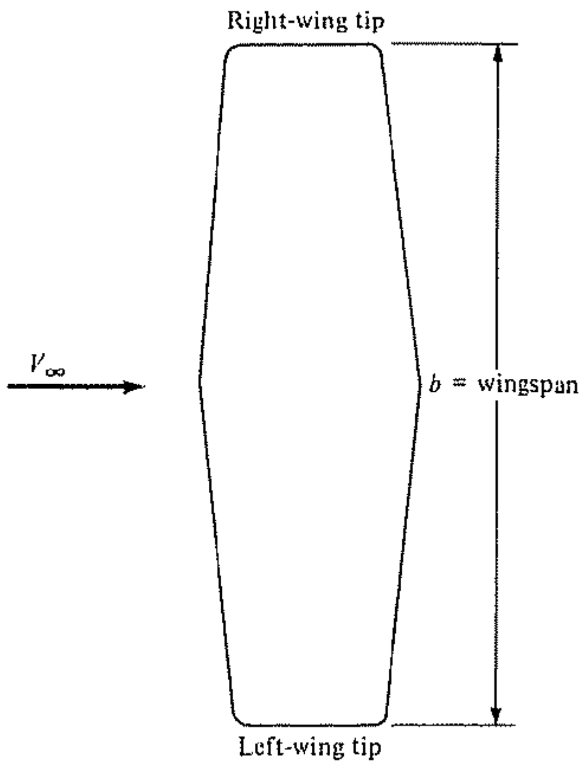


Figure 5.11 Infinite (two-dimensional) wing



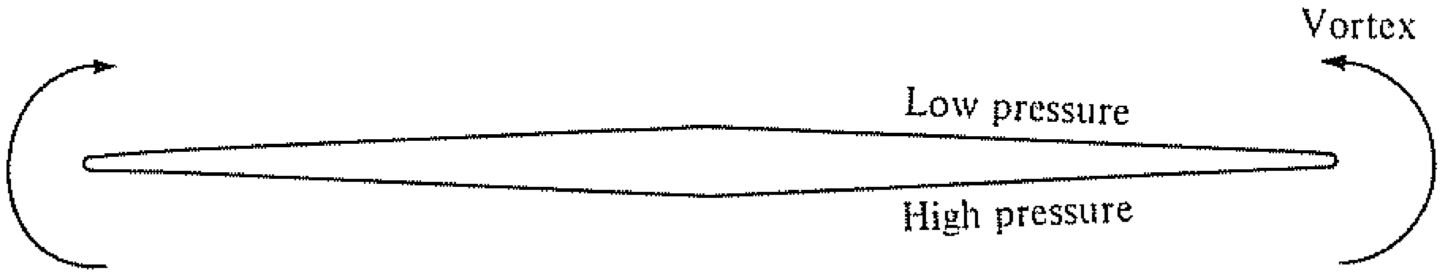
Aspect Ratio
b: wingspan
S: wing area

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

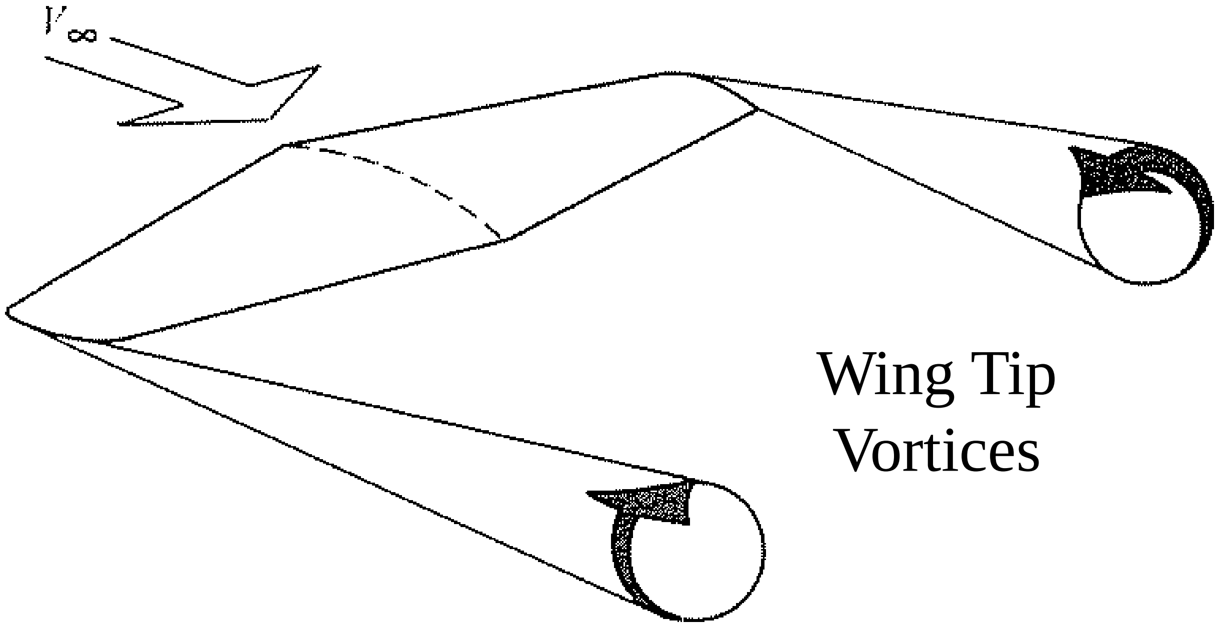
- Upper surface (upper side of wing): low pressure
- Lower surface (underside of wing): high pressure

- Flow always desires to go from high pressure to low pressure

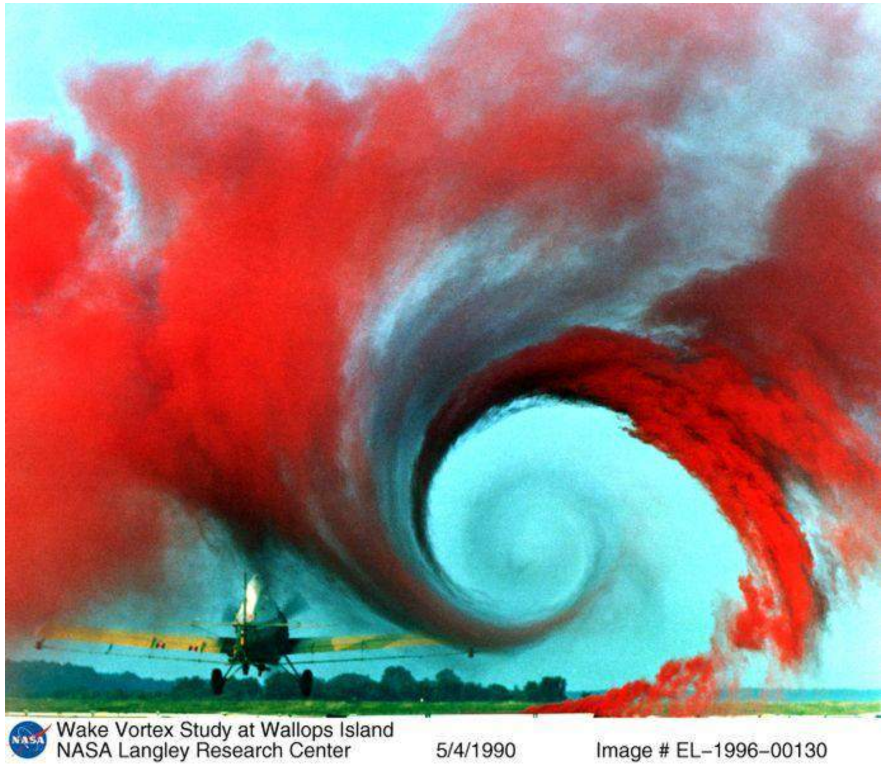
- Flow ‘wraps’ around wing tips



Front View



Wing Tip
Vortices

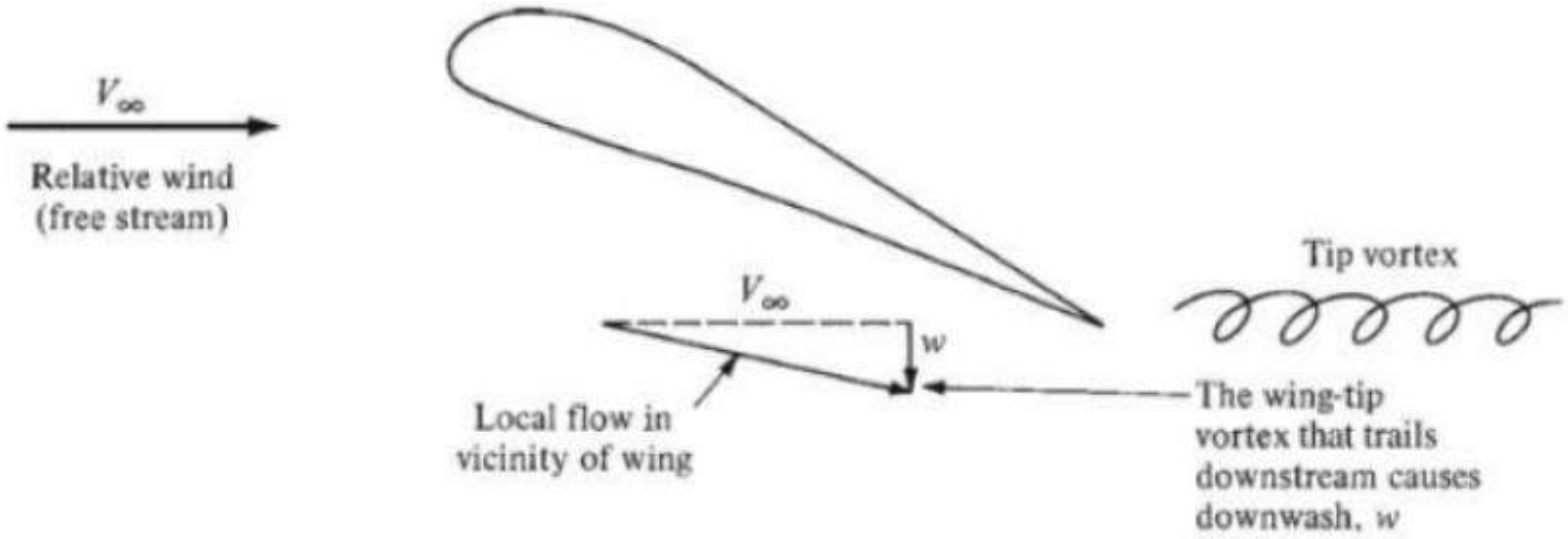

Wake Vortex Study at Wallops Island
NASA Langley Research Center
5/4/1990
Image # EL-1996-00130

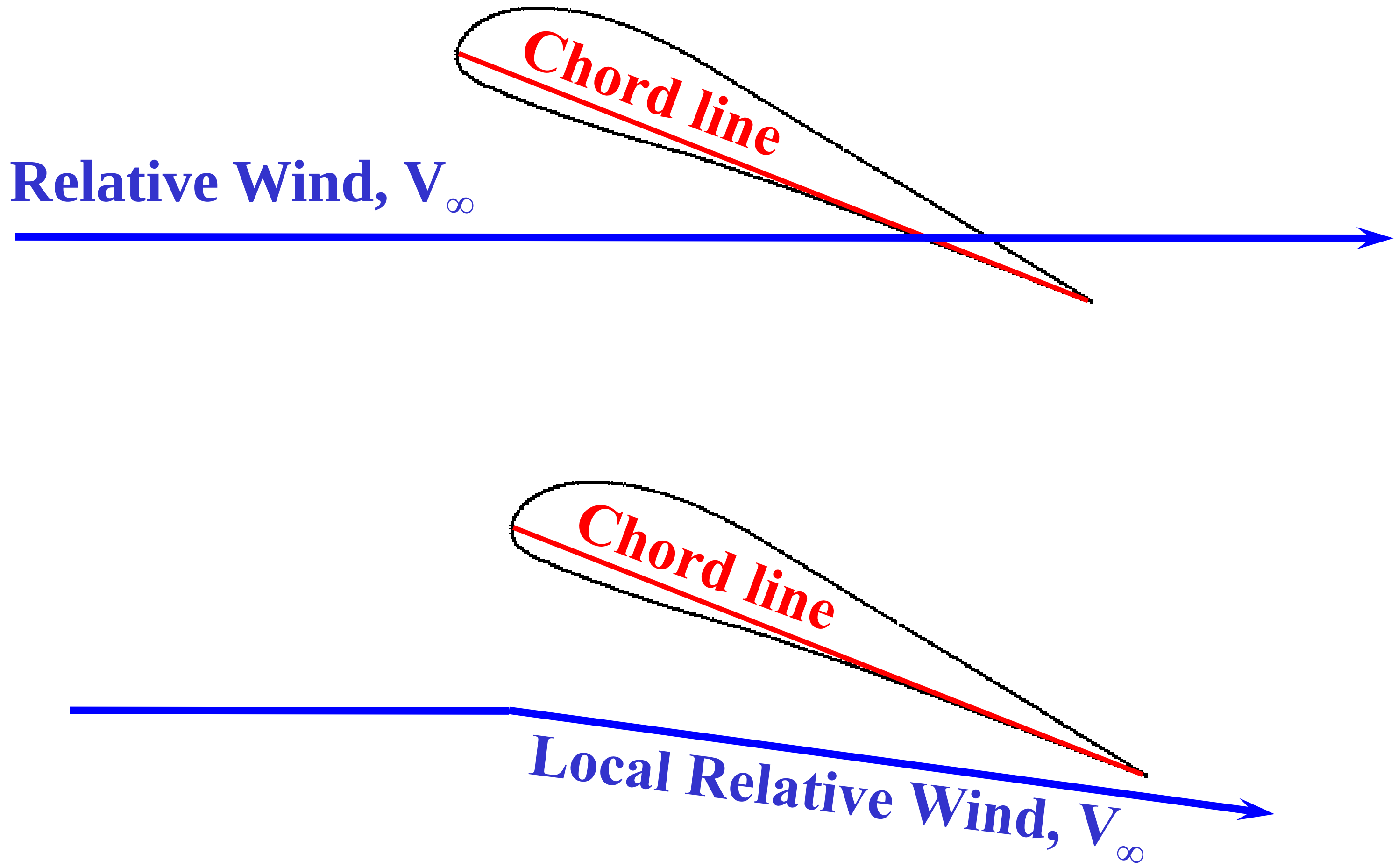
- If the wing has lift, then obviously the average pressure over the bottom surface is greater than that over the top surface.
- Consequently, there is some tendency for the air to "leak," or flow, around the wing tips from the high- to the low-pressure sides.
- This flow establishes a circulatory motion that trails downstream of the wing.
- The trailing circular motion is called a *vortex.*

DOWNWASH

- These wing-tip vortices downstream of the wing induce a small downward component of air velocity in the neighborhood of the wing itself.
- The two wing-tip vortices tend to drag the surrounding air around with them, and this secondary movement induces a small velocity component in the downward direction at the wing.
- This downward component is called *downwash* and given the symbol *w*.

- An effect of downwash can be seen in Fig.
- Vα designates the relative wind.
- In the immediate vicinity of the wing, *V*α and *w* add vectorally to produce a "local" relative wind that is canted downward from the original direction of *V*α




Chord line
Relative Wind, V_∞
Chord line
Local Relative Wind, V_∞

This has several consequences:

1. The angle of attack of the airfoil sections of the wing is varied in comparison to the angle of attack of the wing referenced to $V\alpha$
2. There is an increase in the drag. The increase is called *induced drag,* which has three physical interpretatinos:-

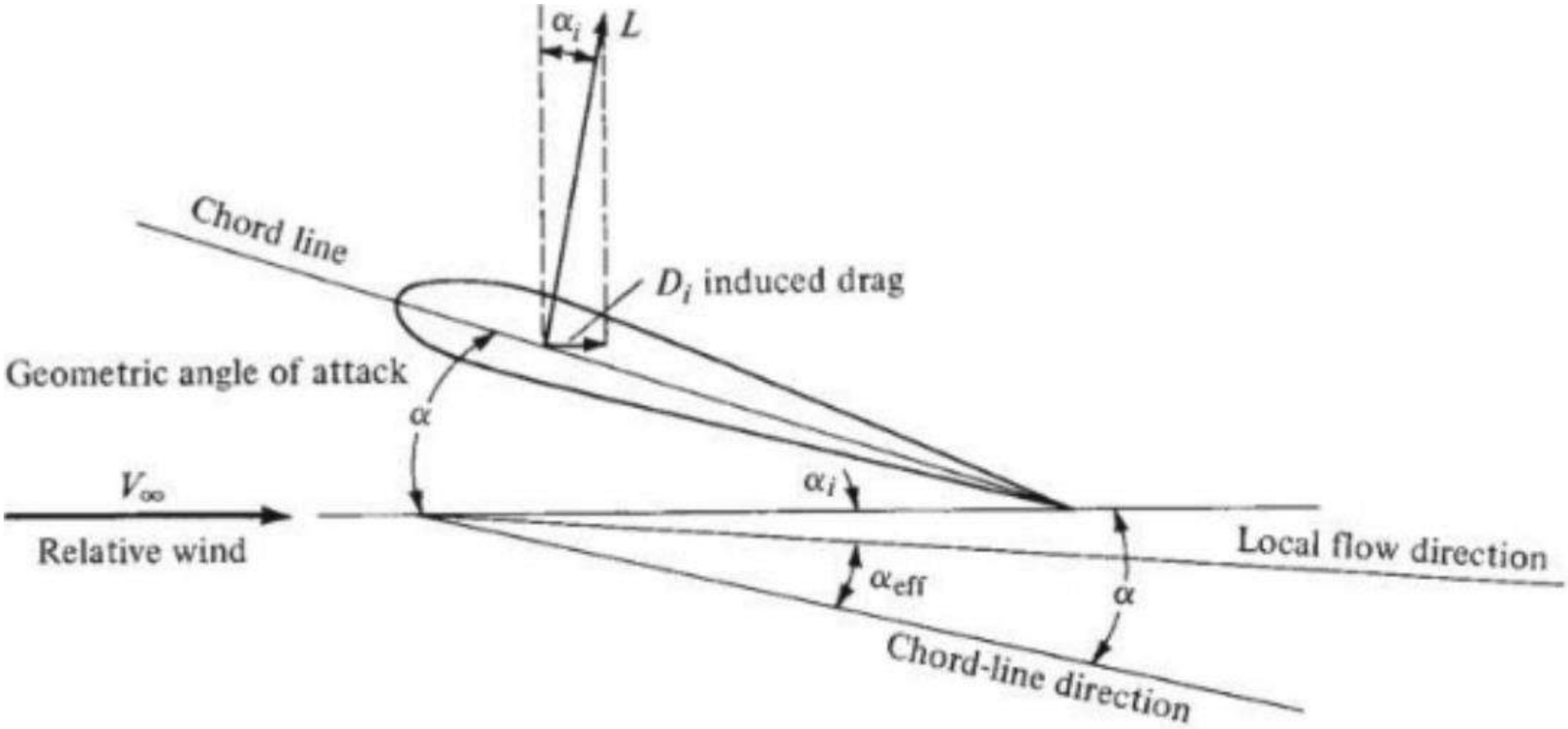
a) First, the wing-tip vortices simply alter the flow field about the wing to change the surface pressure distributions.

b) An alternative explanation is that because the local relative wind is canted downward ,the lift vector itself is "tilted back." Hence, it contributes a certain component of force parallel to $V\alpha$,that is, a drag force.

c) A third physical explanation of the source of induced drag is that the wing-tip vortices contain a certain amount of rotational kinetic energy. This energy has to come from somewhere; it is supplied by the aircraft propulsion system, where extra power has to be added to overcome the extra increment in drag due to induced drag.

Calculation of magnitude of *Di*

- Consider a section of a finite wing as shown in Fig.
- The angle of attack defined between the mean chord of the wing and the direction of $V\alpha$ (the relative wind) is called the *geometric angle of attack* α
- In the vicinity of the wing, the local flow is deflected downward by angle αi because of downwash.
- This angle αi defined as the *induced angle of attack,* is the difference between the local flow direction and the free-stream direction.
- Hence, although the naked eye sees the wing at an angle of attack α ,the airfoil section itself is seeing an *effective angle of attack*, which is smaller than α
- Letting α_{eff} denote the effective angle of attack, from Fig. that $\alpha eff = \alpha - \alpha i$



α_i
L
Chord line
D_i induced drag
Geometric angle of attack
α
V_∞
Relative wind
α_i
Local flow direction
α_{eff}
α
Chord-line direction

$$\boxed{\alpha_{geometric} = \alpha_{effective} + \alpha_{induced}}$$

$\boldsymbol{\alpha}_{\textbf{geometric}}$: what you see, what you would see in a wind tunnel

Simply look at angle between **incoming relative wind** and **chord line**

$\boldsymbol{\alpha}_{\textbf{effective}}$: what the airfoil ‘sees’ locally

Angle between local flow direction and chord line

Small than $\alpha_{\text{geometric}}$ because of downwash

$\boldsymbol{\alpha}_{\textbf{induced}}$: difference between these two angles

Downwash has ‘induced’ this change in angle of attack

- Because the local flow direction in the vicinity of the wing is inclined downward with respect to the free stream, the lift vector remains perpendicular to the local relative wind and is therefore tilted back through angle α_i.
- The tilted-lift vector contributes a certain component of drag. This drag is the *induced drag D_i.*

$$D_i = L \sin \alpha_i$$

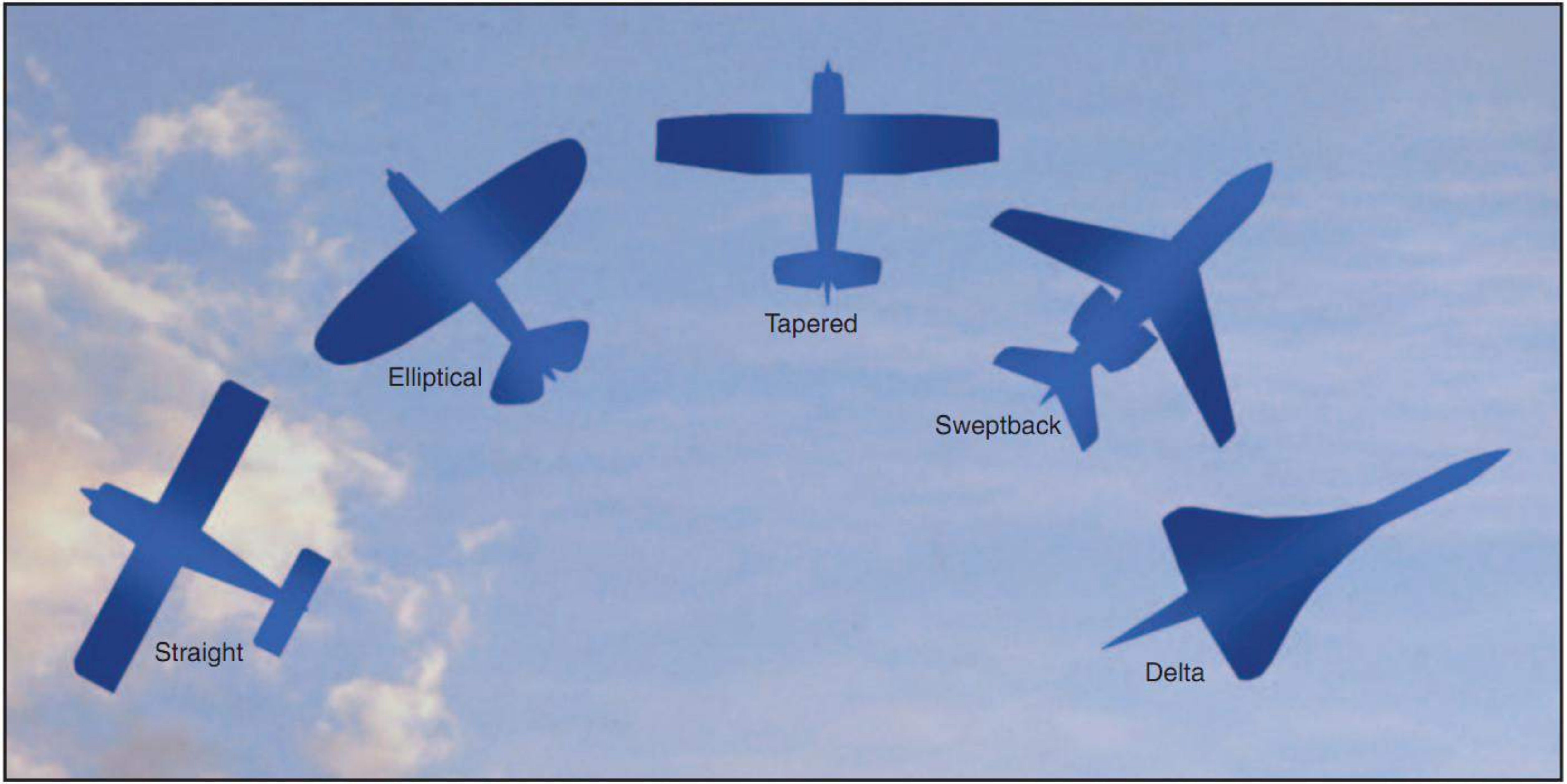
$$\boxed{C_{D,i} = \frac{C_L^2}{\pi \mathrm{AR}}}$$

For all wings in general, a *span efficiency factor e* can be defined such that

$$\boxed{C_{D,i} = \frac{C_L^2}{\pi e \mathrm{AR}}} \tag{5.57}$$

For elliptical planforms, $e = 1$; for all other planforms, $e < 1$. Thus, $C_{D,i}$ and hence induced drag are a *minimum for an elliptical planform*. For typical subsonic aircraft, e ranges from 0.85 to 0.95. Equation (5.57) is an important relation. It demonstrates that induced drag varies as the square of the lift coefficient; at high lift, such as near $C_{L,\max}$, the induced drag can be a substantial portion of the total drag. Equation (5.57) also demonstrates that as AR is increased, induced drag is decreased. Hence, subsonic airplanes designed to minimize induced drag have high–aspect-ratio wings (such as the long, narrow wings of the Lockheed U-2 high-altitude reconnaissance aircraft).

The word planform means shape as view by looking down on the wing



In light of Eq. (5.57), we can now write the total drag coefficient for a finite wing at subsonic speeds as

$$\boxed{\underbrace{C_D}_{\substack{\text{Total} \\ \text{drag}}} = \underbrace{c_d}_{\substack{\text{Profile} \\ \text{drag}}} + \underbrace{\frac{C_L^2}{\pi e \text{AR}}}_{\substack{\text{Induced} \\ \text{drag}}}} \tag{5.58}$$

TOTAL DRAG ON SUBSONIC WING

$$D = D_{friction} + D_{pressure} + D_{induced}$$

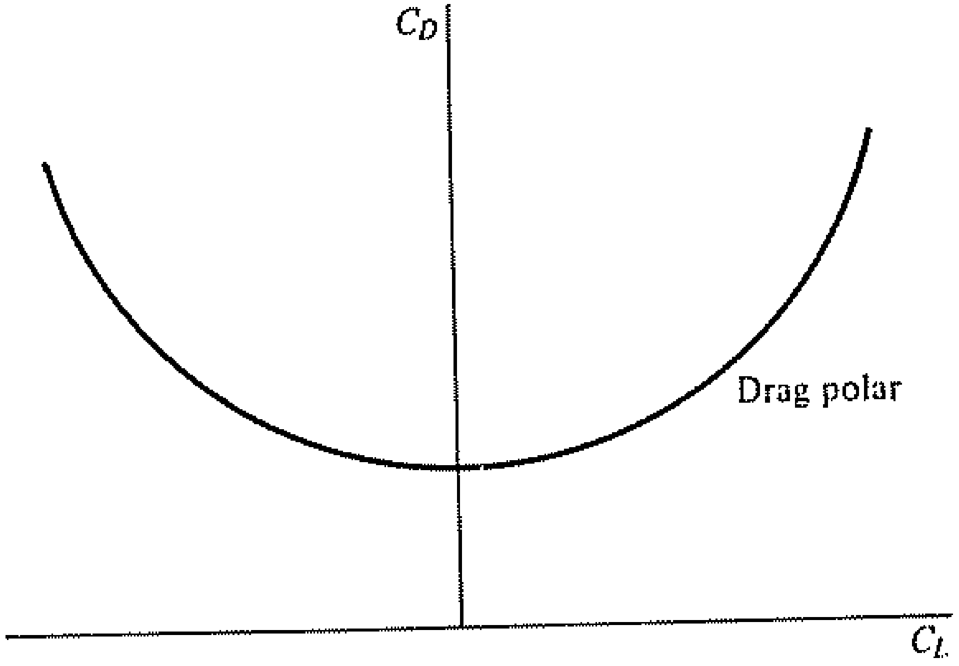
$$D = D_{profile} + D_{induced}$$

DRAG POLAR

- The quadratic variation of C_D with C_L, when plotted on a graph, leads to a curve as shown in Fig.
- Such a plot of C_D versus C_L is called a *drag polar.*
- Much of the basic aerodynamics of an airplane is reflected in the drag polar, and such curves are essential to the design of airplanes.

$$\boxed{C_D} = \boxed{c_d} + \boxed{\frac{C_L^2}{\pi e AR}}$$

$$\boxed{\text{Total Drag}} = \boxed{\text{Profile Drag}} + \boxed{\text{Induced Drag}}$$



- For an elliptical lift distribution, gives values for the induced angle of attack

$$\alpha_i = \frac{C_L}{\pi \mathrm{AR}} \tag{5.53}$$

- Extending Eq. (5.53) to wings of any general planform, we can define a new span effectiveness factor e_1 such that:

$$\alpha_i = \frac{C_L}{\pi e_1 \mathrm{AR}}$$ In radians

$$\alpha_i = \frac{57.3 C_L}{\pi e_1 \mathrm{AR}}$$ In degrees

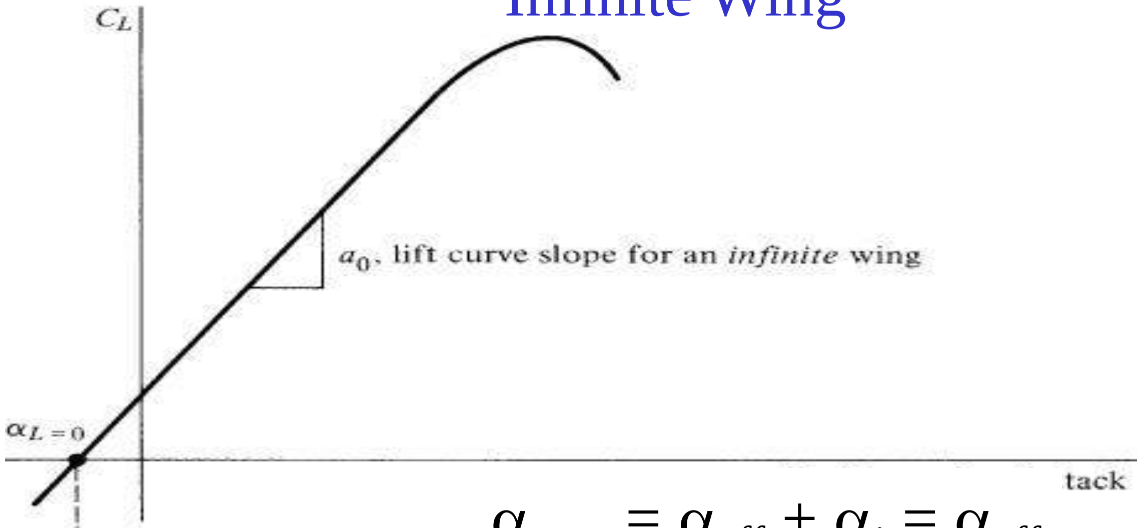
CHANGE IN THE LIFT SLOPE

- The aerodynamic properties of a finite wing differ in two major respects from infinite wings.

1) the addition of induced drag for a finite wing.

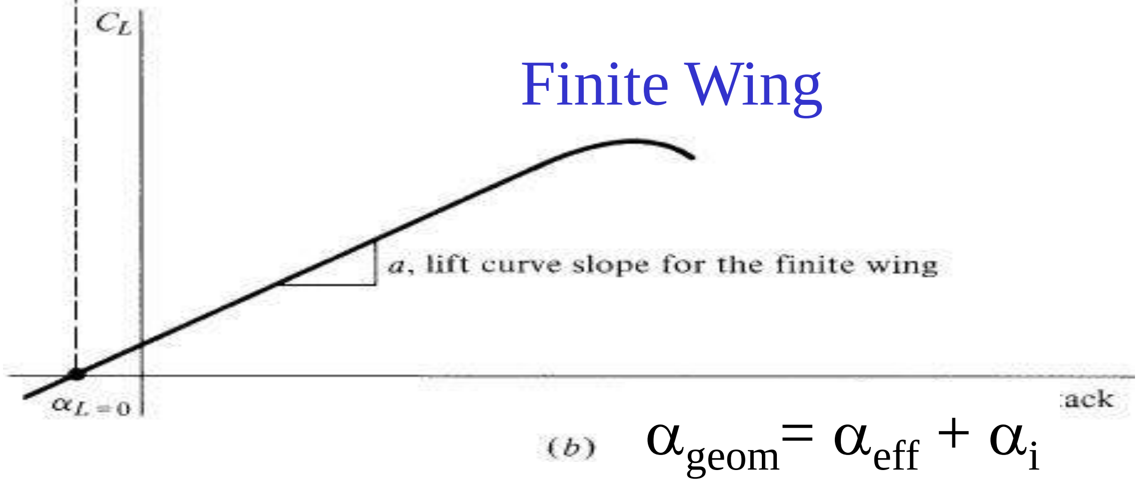
2) The lift curve for a finite wing has a smaller slope than the corresponding lift curve for an infinite wing with the same airfoil cross section.

Infinite Wing



$\alpha_{geom} = \alpha_{eff} + \alpha_i = \alpha_{eff}$

Finite Wing



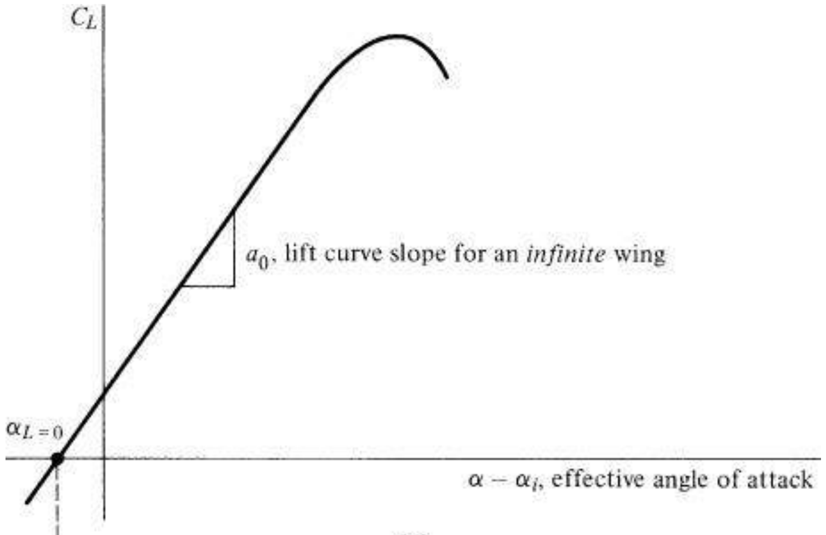
$\alpha_{geom} = \alpha_{eff} + \alpha_i$

- For infinite wings, there is no induced angle of attack

- With finite wings, there is an induced angle of attack

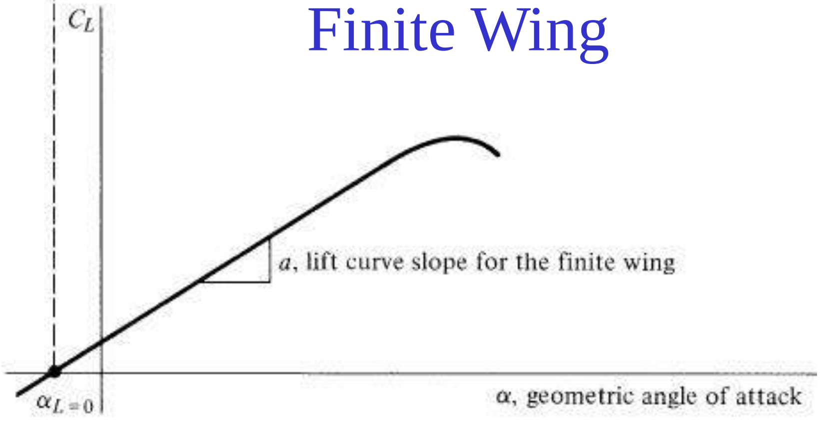
$$\boxed{\alpha_{geom} = \alpha_{eff} + \alpha_i}$$

Infinite Wing



(a)

Finite Wing



(b)

- **Lift curve for a finite wing has a smaller slope than corresponding curve for an infinite wing with same airfoil cross-section**
  - Figure (a) shows infinite wing, $\alpha_i = 0$, so plot is C_L vs. α_{geom} or α_{eff} and slope is a_0
  - Figure (b) shows finite wing, $\alpha_i \neq 0$
    - Plot C_L vs. α_{geom}

Effective of finite wing is to reduce lift curve slope

- Finite wing lift slope = α = $dC_L/d\alpha$

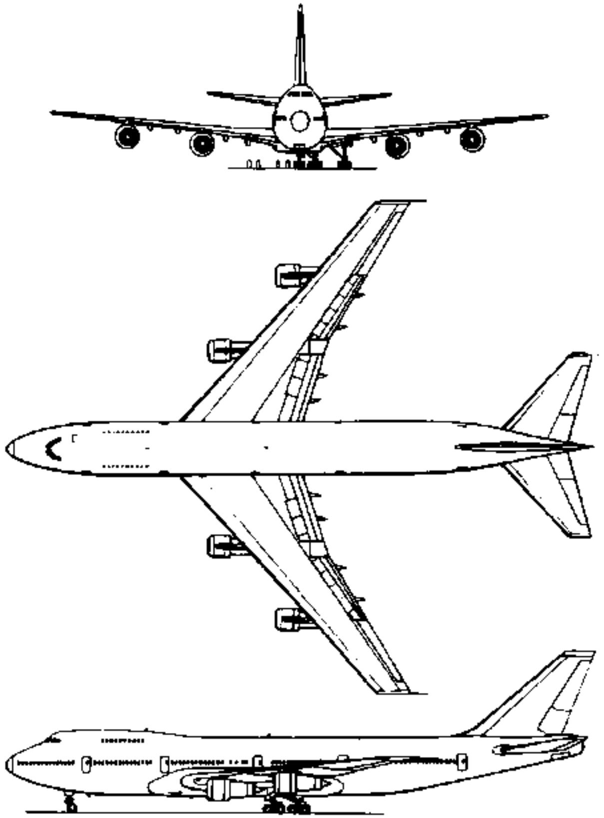
SUMMARY

- **$C_{D,i}$ proportional to $C_L{}^2$**
  - Airplane on take-off or landing, induced drag is a major component
  - Significant at cruise (15-25% of total drag)

- **$C_{D,i}$ inversely proportional to AR**
  - Desire high AR to reduce induced drag
  - Compromise between structures and aerodynamics
  - AR important tool as designer (more control than span efficiency, e)

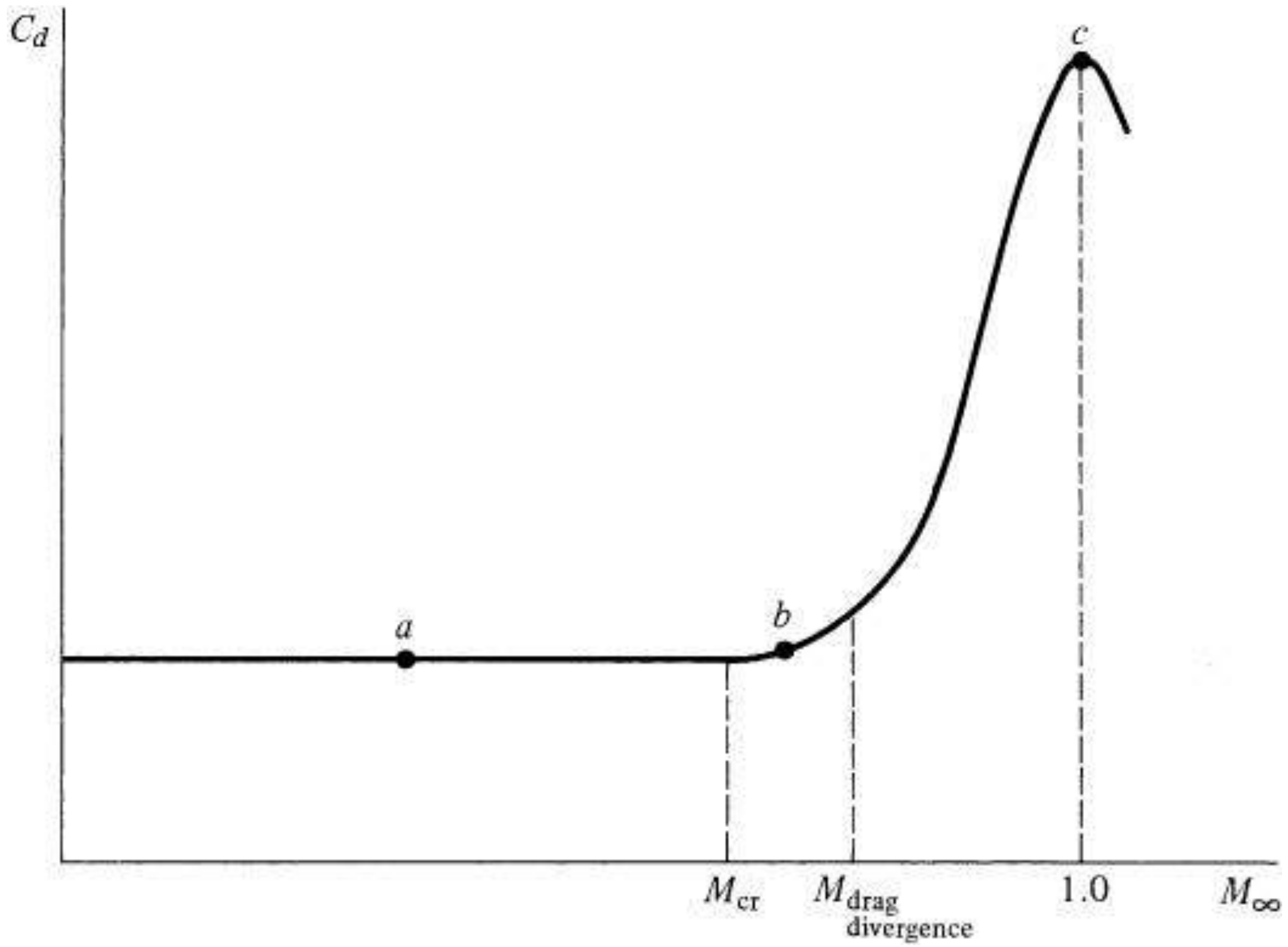
SWEPT WINGS

SWEPT WINGS

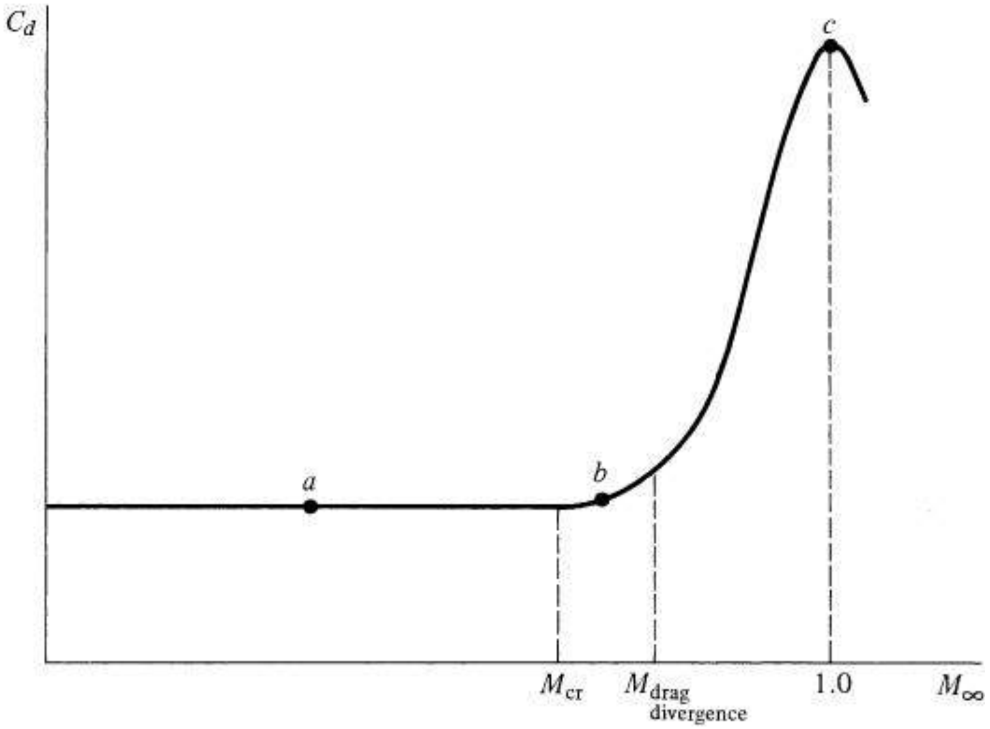


DRAG-DIVERGENCE MACH NUMBER

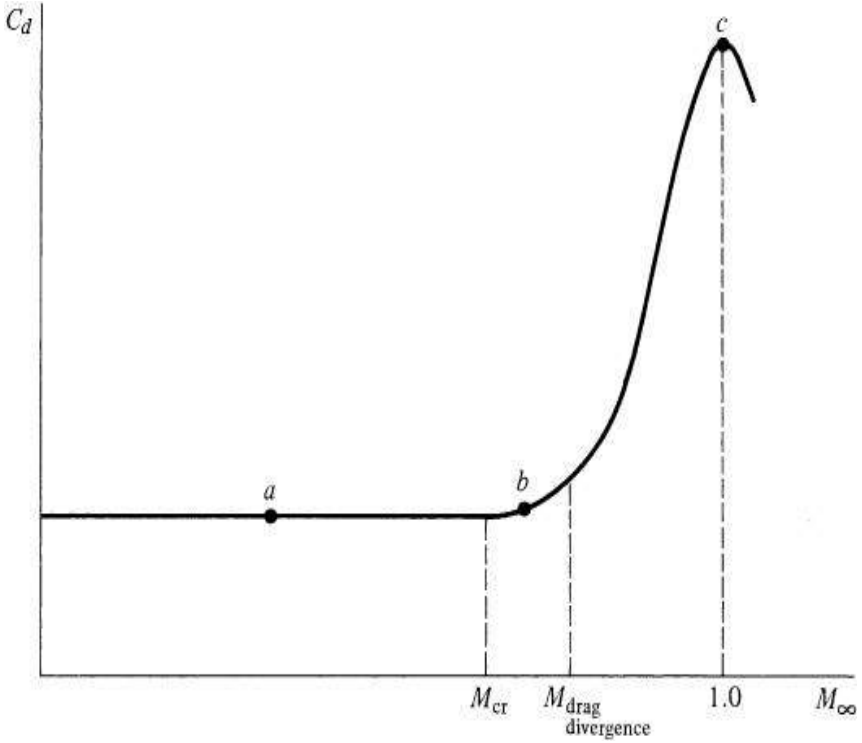
- Figure shows the variation of *cd* with Mα
- At low Mach numbers, less than *Mcr* , *cd* is virtually constant and is equal to its low-speed value

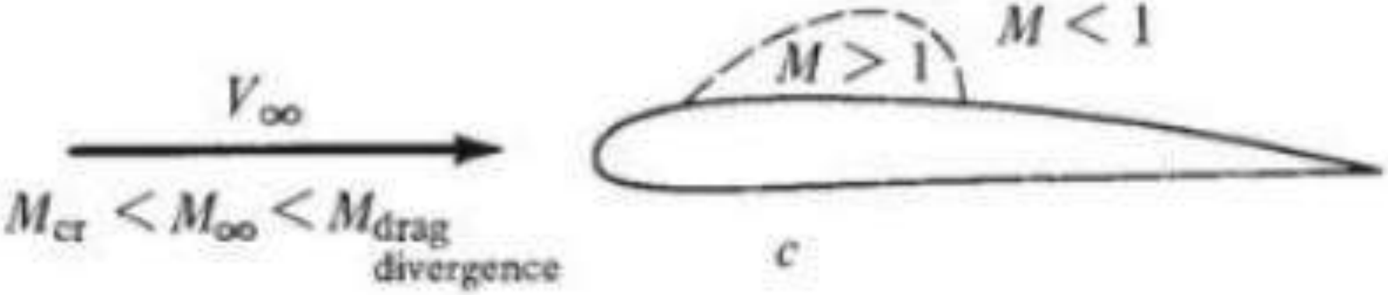


- The flow field about the airfoil for this condition is noted in Fig. *a*, where $M < 1$ everywhere in the flow.



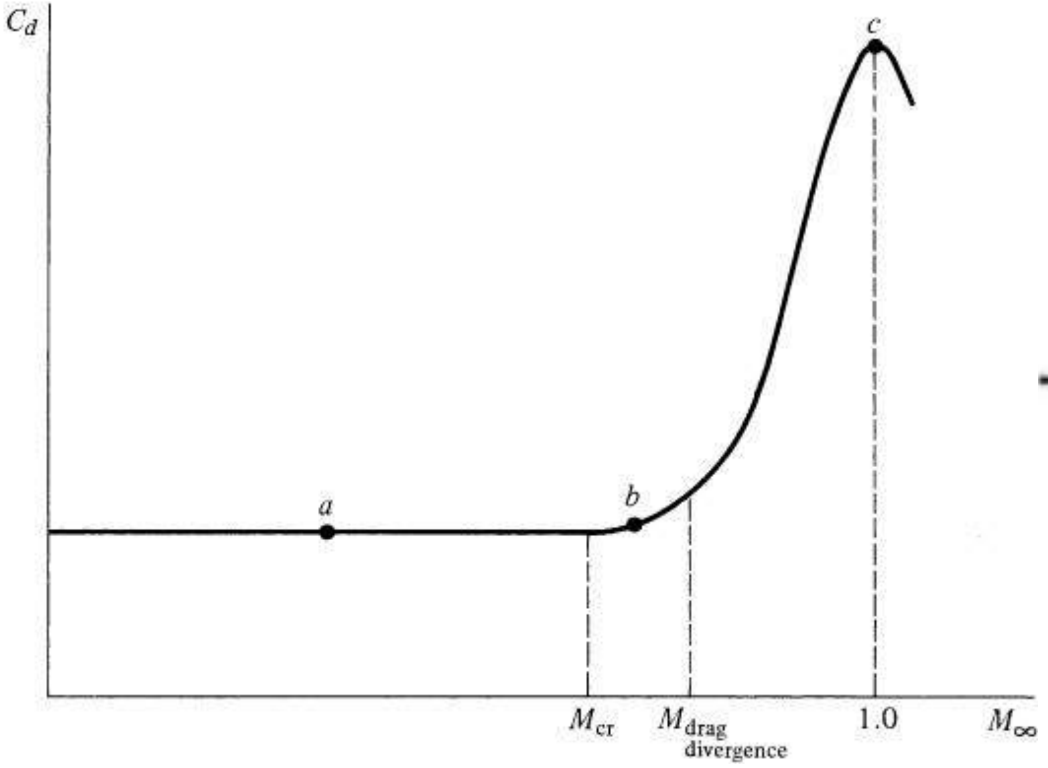
(*a*)

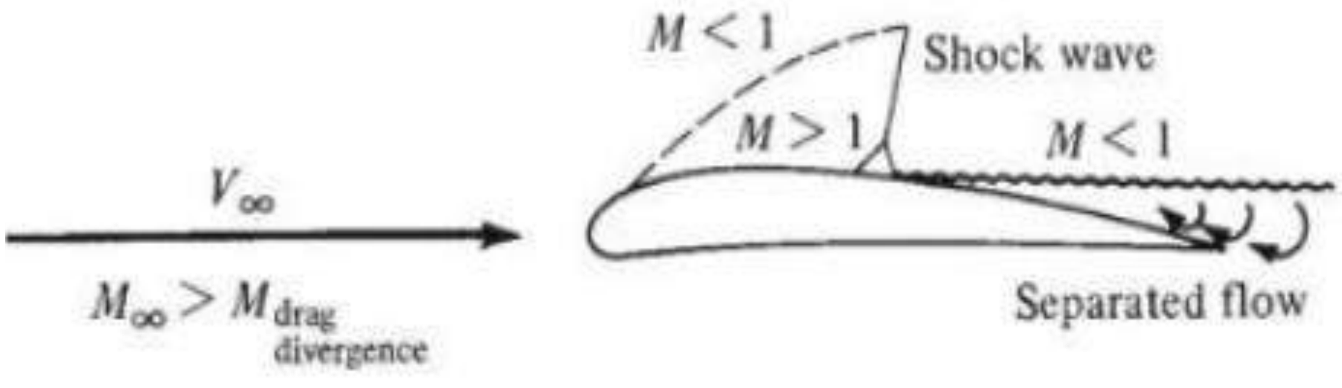
- If Mα is increased slightly above *Mcr* a "bubble" of supersonic flow will occur, surrounding the minimum pressure point, as shown in Fig. *b*.
- Correspondingly, *cd* will still remain reasonably low, as indicated by point *b* in Fig.
- **Modern airfoils and airplanes operate near point b**
- 



(b)

- However, if Mα is still further increased, a very sudden and dramatic rise in the drag coefficient will be observed, as noted by point *c* in Fig..
- Here shock waves suddenly appear in the flow, as sketched in Fig. c.





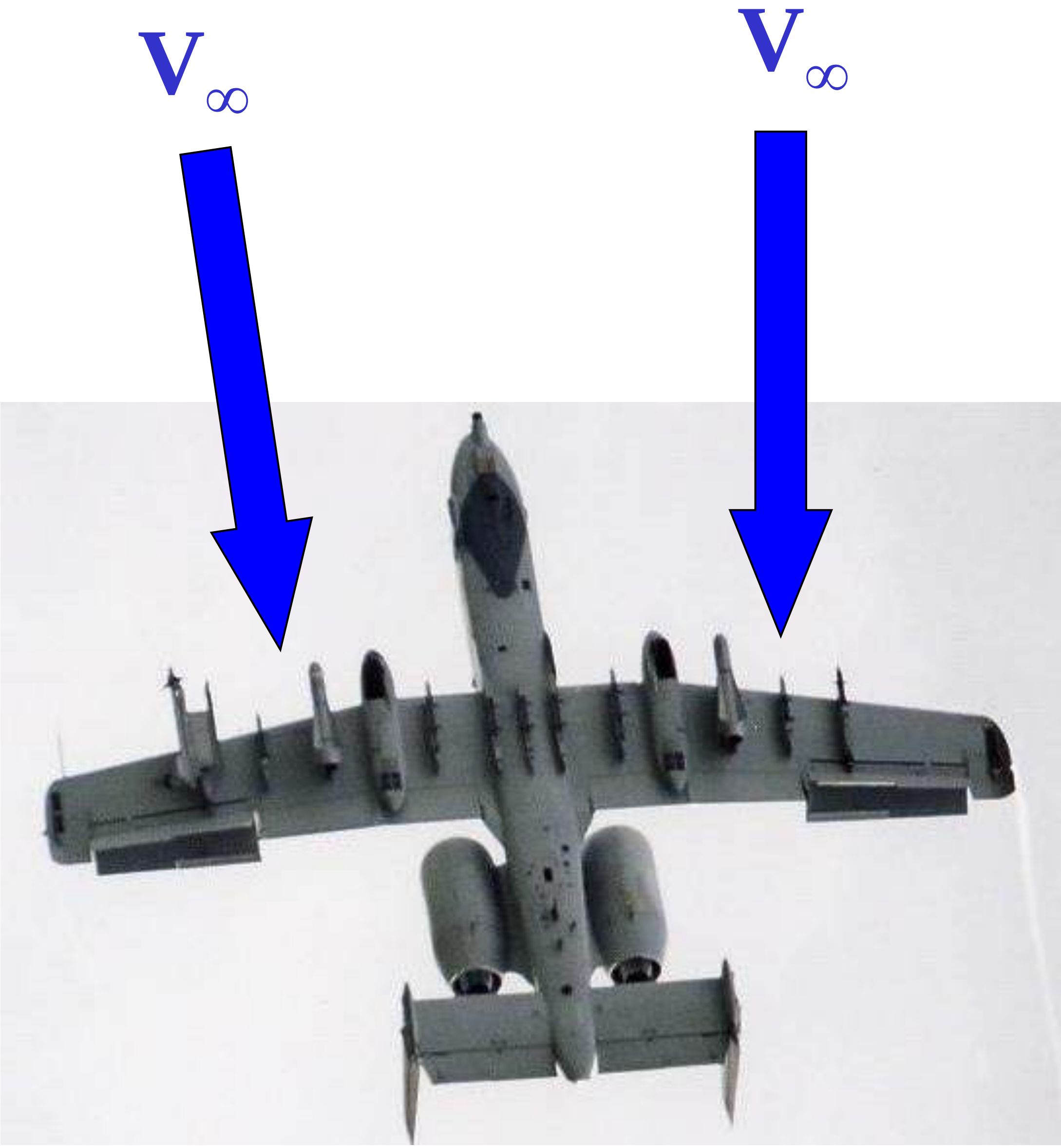
(*c*)

- The shock waves themselves are dissipative phenomena that increase drag on the airfoil. But in addition, the sharp pressure increase across the shock waves creates a strong adverse pressure gradient, causing the flow to separate from the surface.
- Such flow separation can create substantial increases in drag. Thus, the sharp increase in *cd* is a combined effect of shock waves and flow separation. The *free-stream* Mach number at which *cd* begins to increase rapidly is defined as the *drag-divergence Mach number*

$$M_{CR} < M_{Drag-Divergence} < 1.0$$

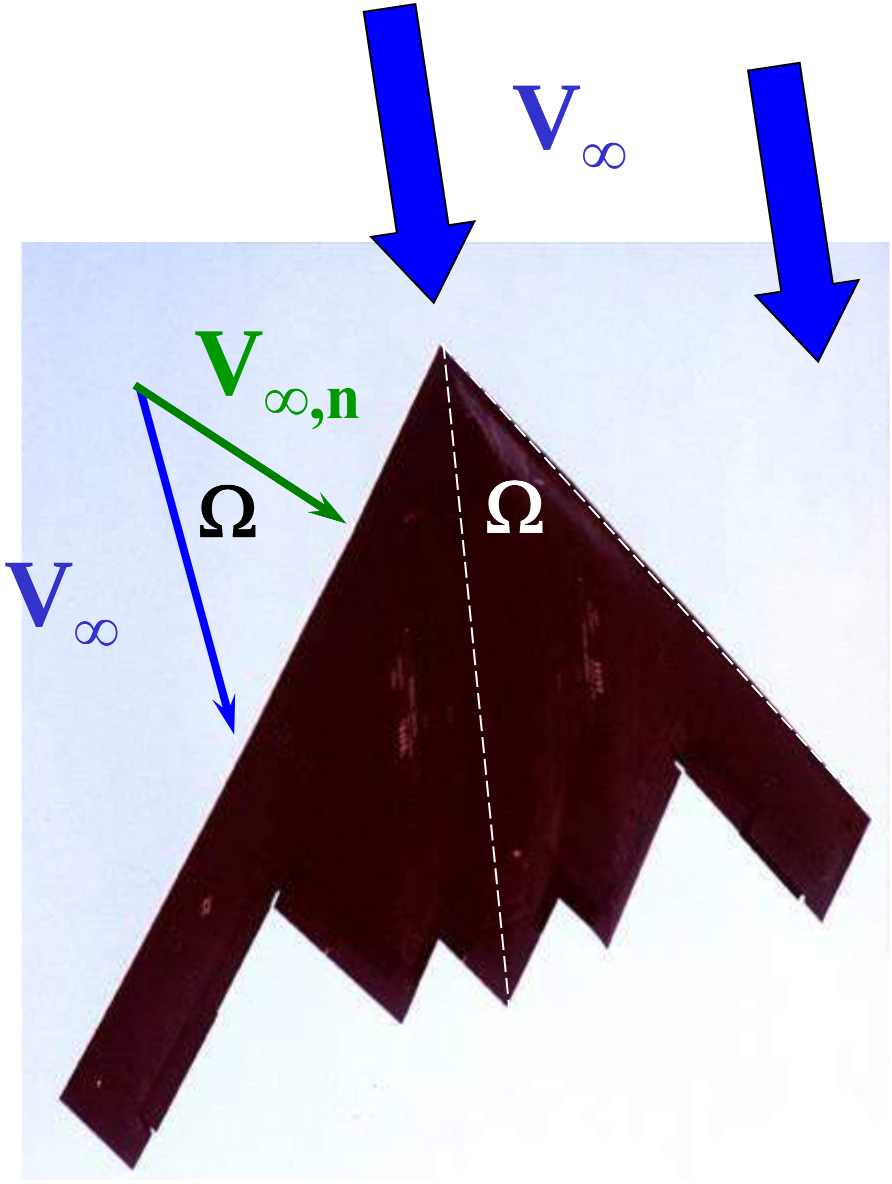
- The shock pattern sketched in Fig. c is characteristic of a flight regime called *transonic*.
- When $0.8 \leq M\alpha \leq 1.2$, the flow is generally designated as transonic flow.

WHY WING SWEEP?



Wing sees component of flow <u>normal</u> to leading edge

WHY WING SWEEP?

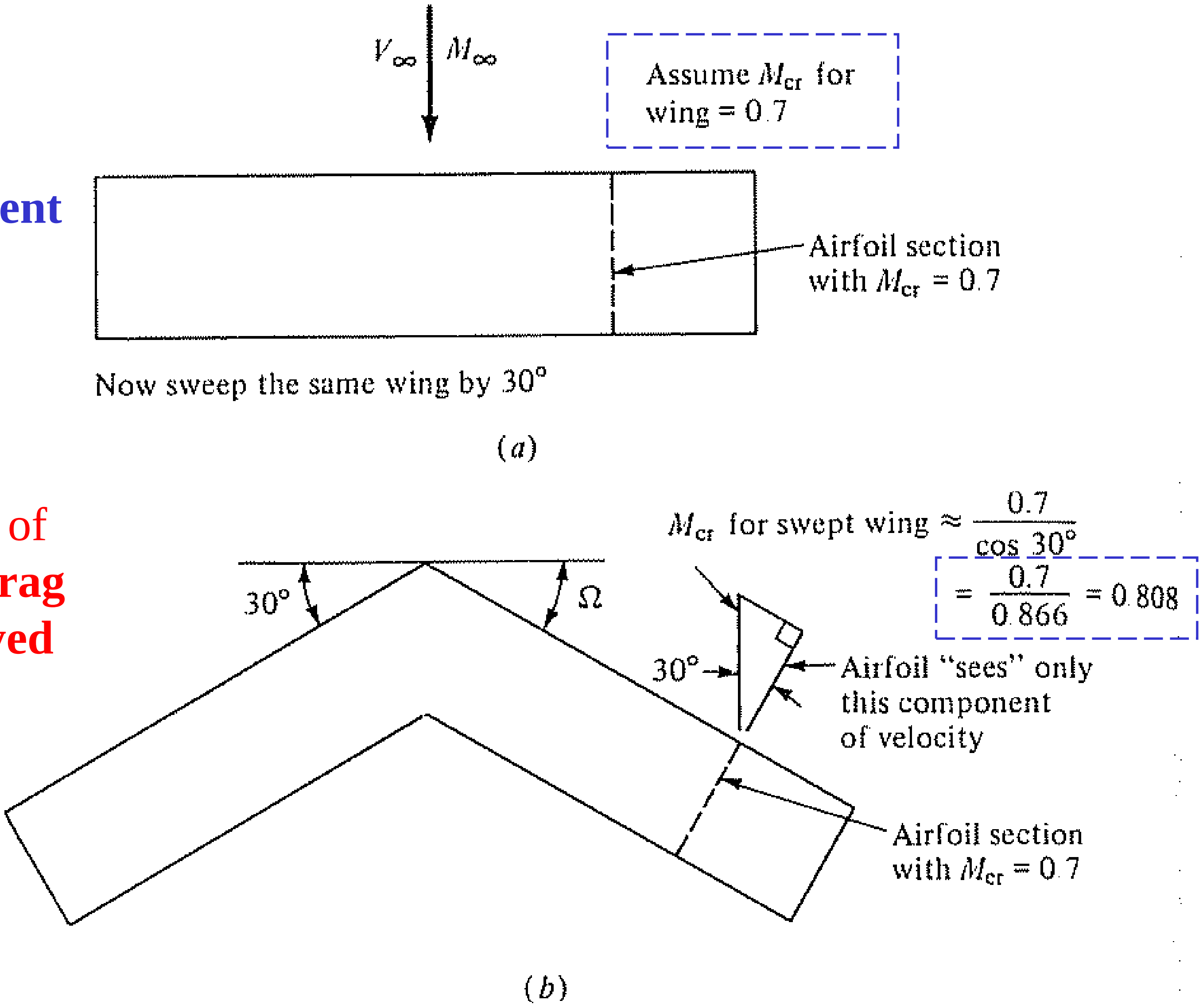


$V_{\infty,n} < V_\infty$

Wing sees component of flow <u>normal</u> to leading edge

SWEPT WINGS: SUBSONIC FLIGHT

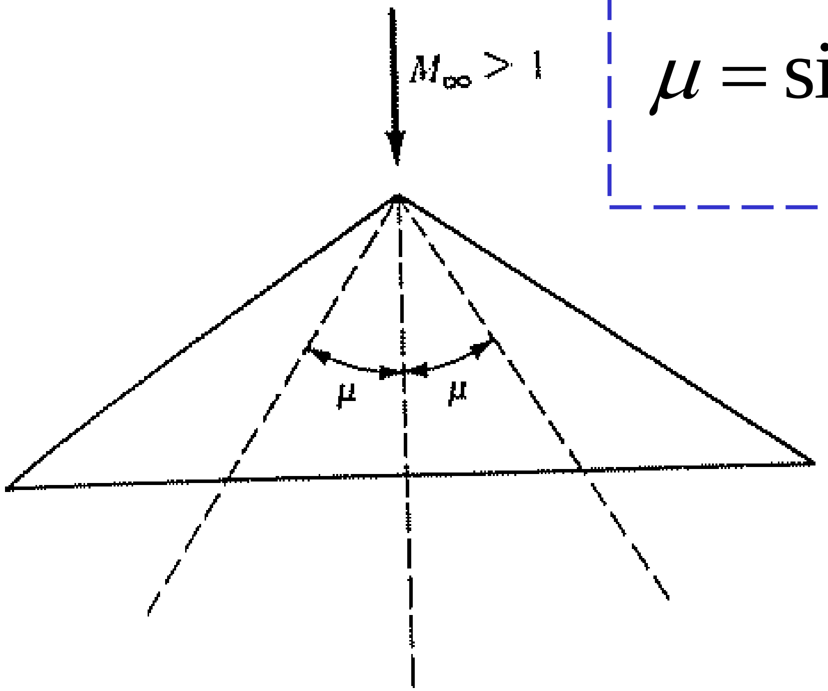
- Recall M_{CR}
- If $M_\infty > M_{CR}$ large increase in drag
- **Wing sees component of flow <u>normal</u> to leading edge**
- Can increase M_∞
- By sweeping wings of **<u>subsonic</u>** aircraft, **drag divergence is delayed to higher Mach numbers**



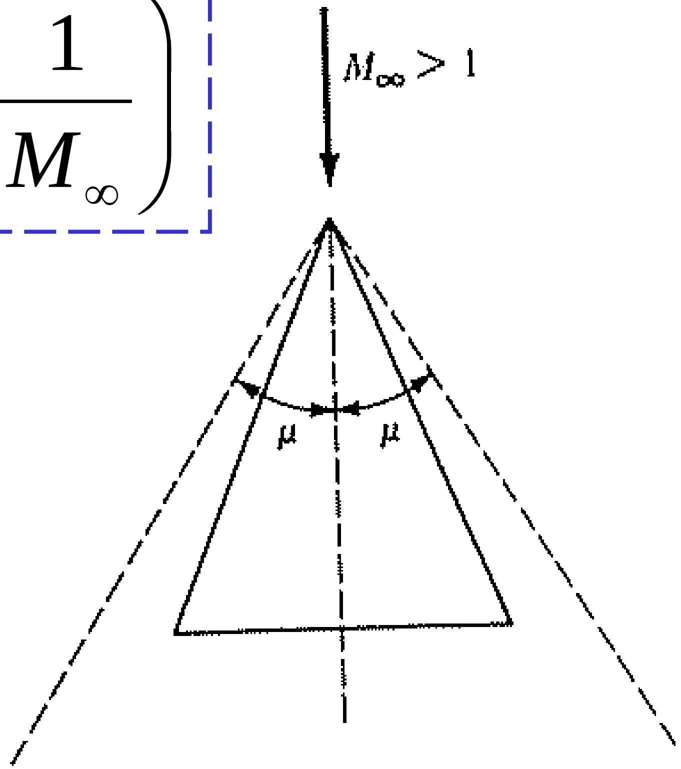
SWEPT WINGS: SUPERSONIC FLIGHT

- If the leading edge of a swept wing is *outside* the Mach cone, as shown in Fig. a, the component of the Mach number normal to the leading edge is supersonic.
- As a result, a fairly strong oblique shock wave will be created by the wing itself, with an attendant large wave drag.
- In contrast, if the leading edge of the swept wing is *inside* the Mach cone, as shown in Fig. *b*, the component of the Mach number normal to the leading edge is subsonic.
- As a result, the wave drag produced by the wing is less.
- Therefore, the advantage of sweeping the wings for supersonic flight is in general to obtain a decrease in wave drag; and if the wing is swept inside the Mach cone, a considerable decrease can be obtained

SWEPT WINGS: SUPERSONIC FLIGHT



$$\mu = \sin^{-1}\left(\frac{1}{M_\infty}\right)$$



- If leading edge of swept wing is outside Mach cone, component of Mach number normal to leading edge is supersonic → Large Wave Drag
- If leading edge of swept wing is inside Mach cone, component of Mach number normal to leading edge is subsonic → Reduced Wave Drag
- For supersonic flight, swept wings reduce wave drag

FLAPS : A MECHANISM FOR HIGH LIFT

The slowest speed at which an airplane can fly in straight and level flight is defined as the *stalling speed* , V_{stall}.

The calculation of V_{stall}, as well as aerodynamic methods of making V_{stall} as small as possible, is of vital importance.

The stalling velocity is readily obtained in terms of the maximum lift coefficient. From the definition of C_L,

$$L = q_\infty S C_L = \tfrac{1}{2}\rho_\infty V_\infty^2 S C_L$$

Thus

$$V_\infty = \sqrt{\frac{2L}{\rho_\infty S C_L}} \tag{5.69}$$

In steady, level flight, the lift is just sufficient to support the weight W of the aircraft; that is, $L = W$. Thus

$$V_\infty = \sqrt{\frac{2W}{\rho_\infty S C_L}} \tag{5.70}$$

$$\boxed{V_{\text{stall}} = \sqrt{\frac{2W}{\rho_\infty S C_{L,\max}}}}$$

- To decrease *Vstall*, CL,max must be increased.
- However, for a wing with a given airfoil shape, CL,max is fixed by nature; that is, the lift properties of an airfoil, including maximum lift, depend on the physics of the flow over the airfoil.
- The lifting properties of a given airfoil can be greatly enhanced by the use of"artificial" high-lift devices.
- The most common of these devices is the flap at the trailing edge of the wing, as sketched in Fig

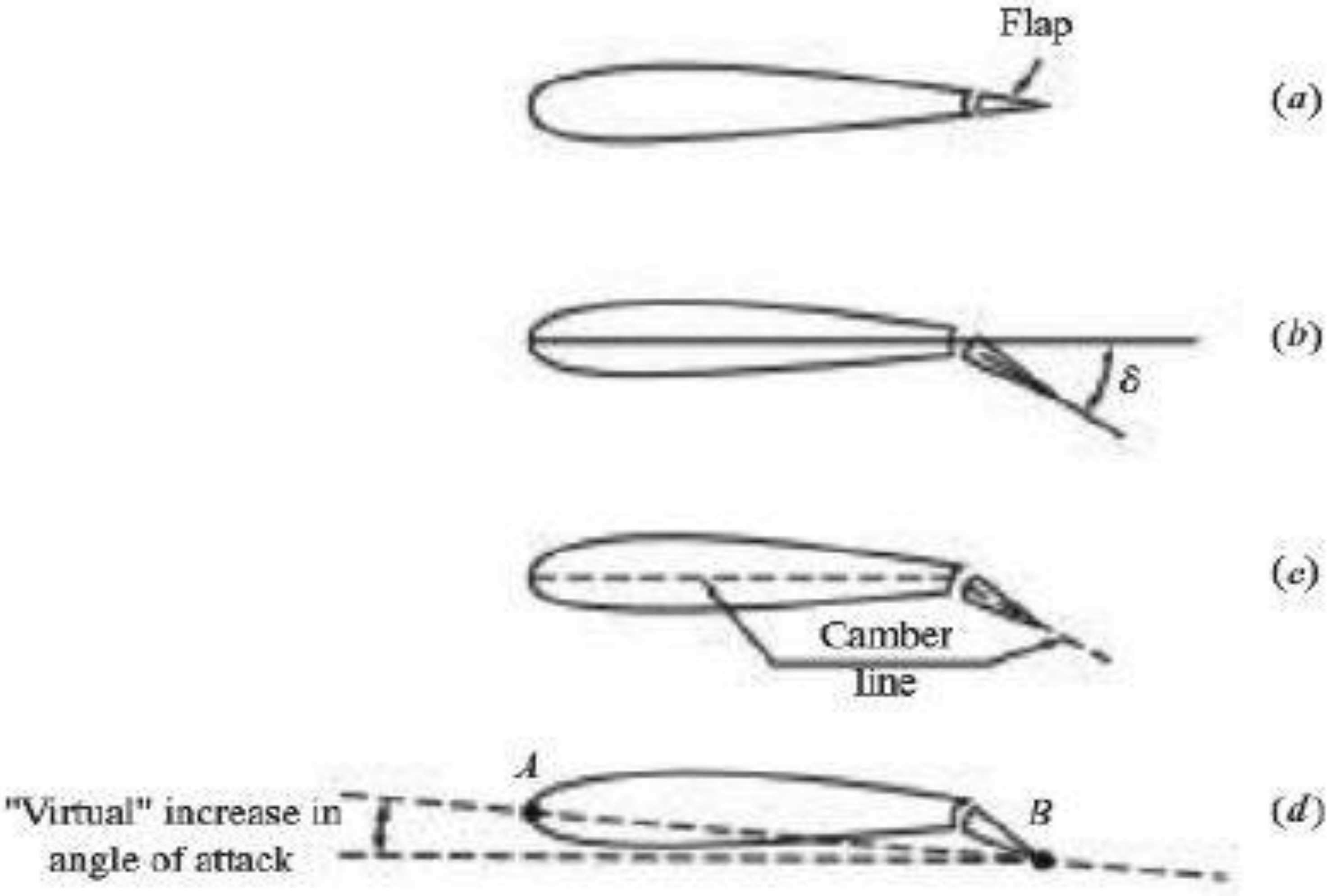


Figure 5.67 When a plain flap is deflected, the increase in lift is due to an effective increase in camber and a virtual increase in angle of attack.

- When the flap is deflected downward through the angle Θ, as sketched in Fig, the lift coefficient is increased for the following reasons:

  1. The camber of the airfoil section is effectively increased, as sketched in Fig.c. The more camber an airfoil shape has at a given angle of attack , the higher the lift coefficient.

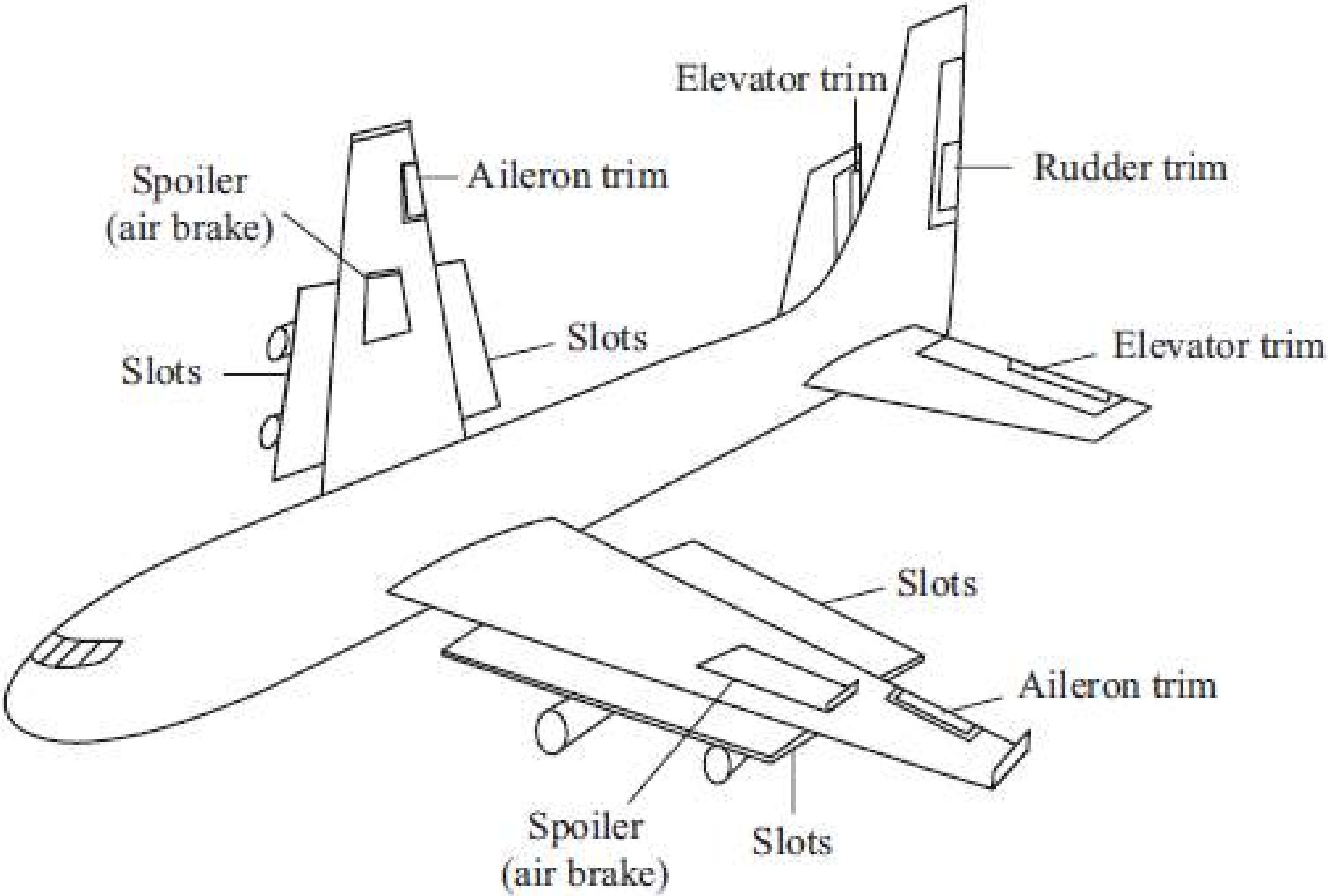
  2. When the flap is deflected, we can visualize a line connecting the leading edge of the airfoil and the trailing edge of the flap: points *A* and B, respectively, in Fig. d.

  Line *AB* constitutes a *virtual chord line*, rotated clockwise relative to the actual chord line of the airfoil, making the airfoil section with the deflected flap see a "virtual" increase in angle of attack. Hence the lift coefficient is increased.

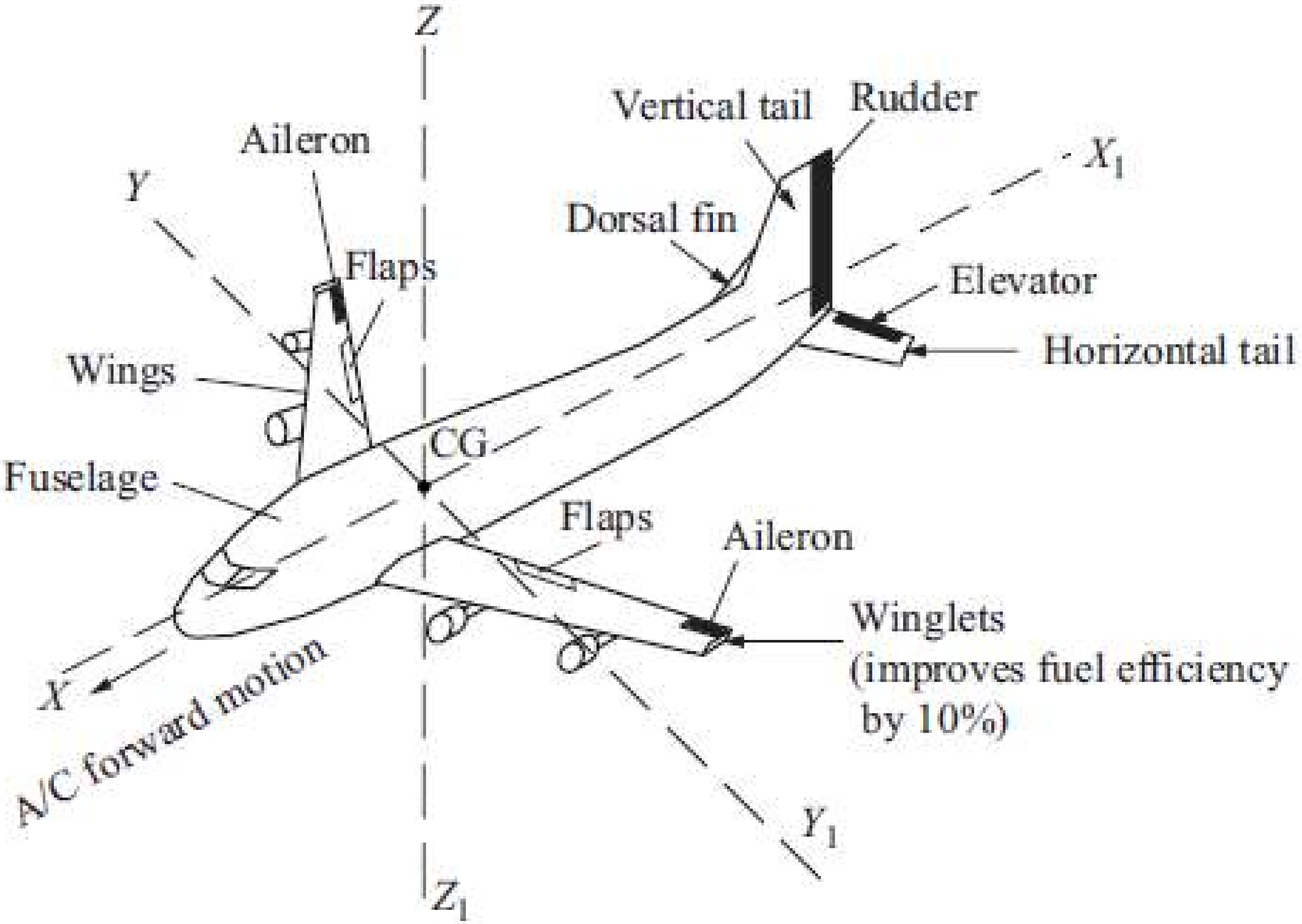
Control Surfaces

An aircraft has two types of control surface:

1. Primary control surfaces and
2. Secondary control surfaces



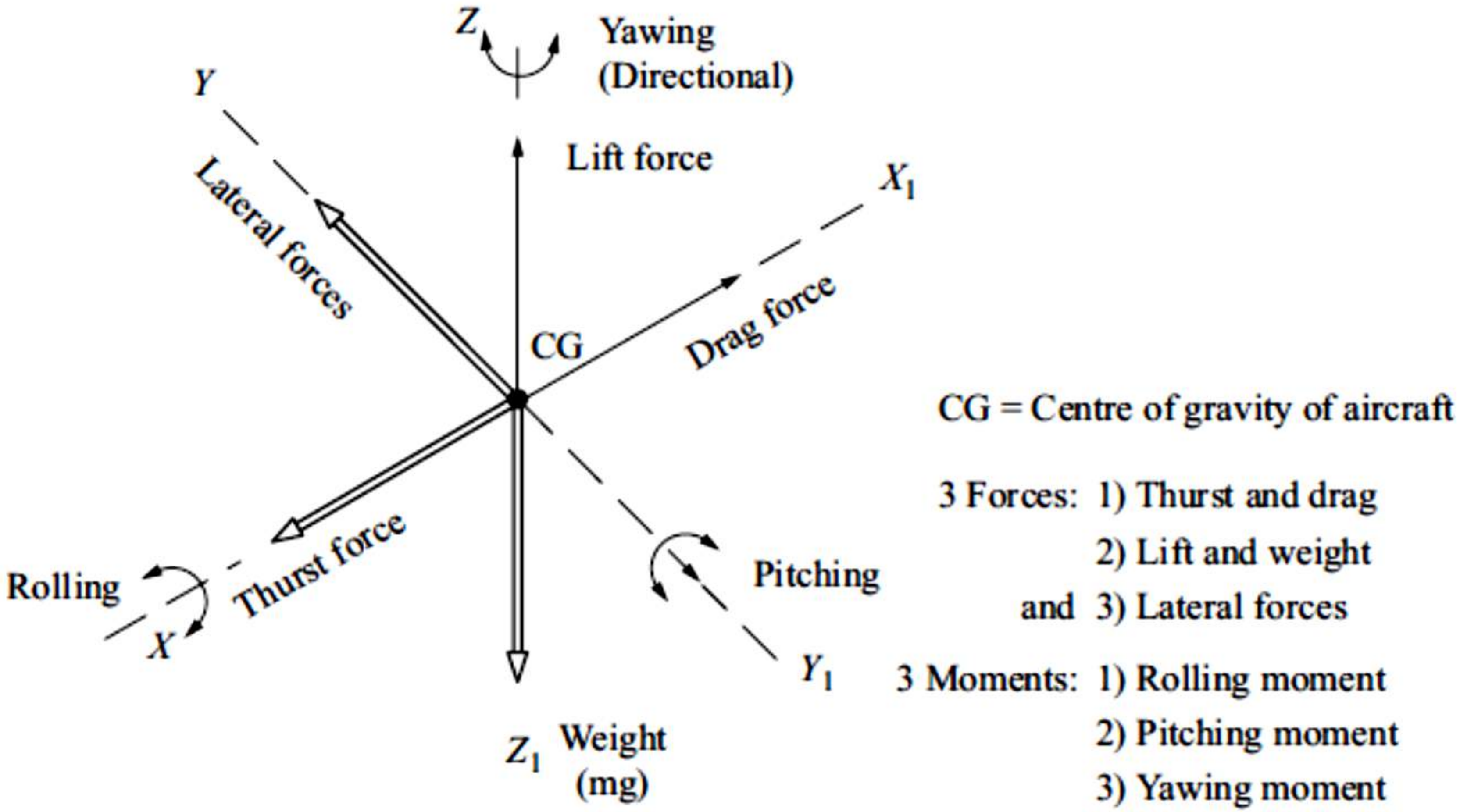
Elevator trim
Spoiler
(air brake)
Aileron trim
Rudder trim
Slots
Slots
Elevator trim
Slots
Aileron trim
Spoiler
(air brake)
Slots



Z
Vertical tail
Rudder
Aileron
X_1
Y
Dorsal fin
Flaps
Elevator
Horizontal tail
Wings
CG
Fuselage
Flaps
Aileron
Winglets
(improves fuel efficiency
by 10%)
X
A/C forward motion
Y_1
Z_1

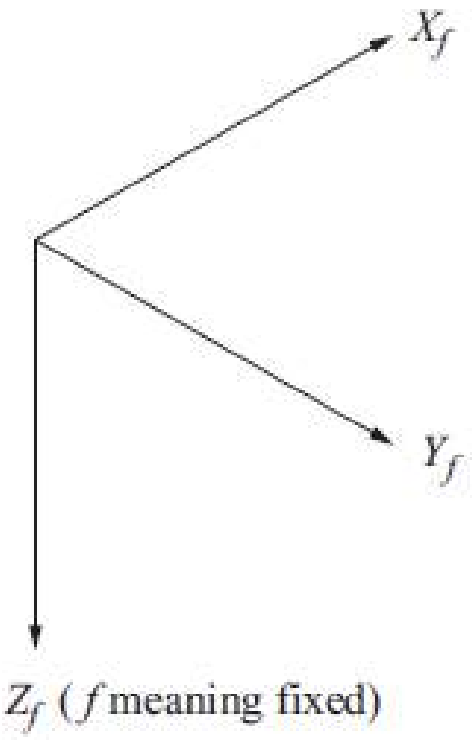
Forces, Moments and Angle of Attack (AOA)

- There are three forces and three moments.



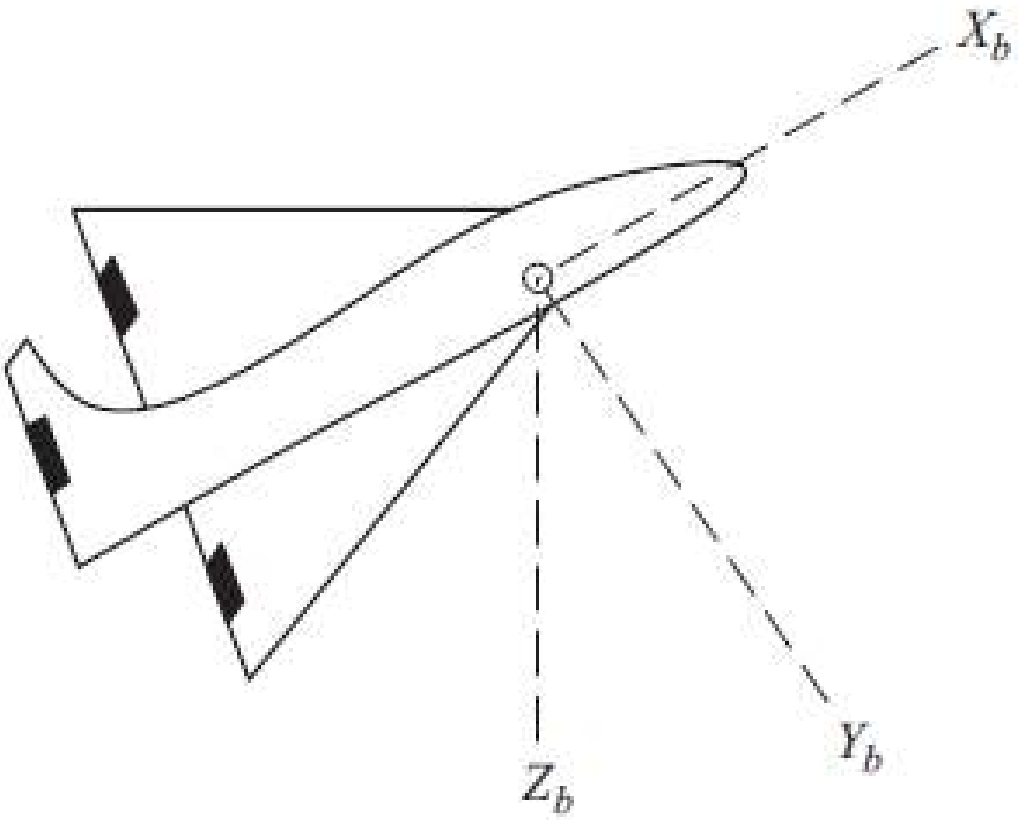
In order to deal with the motion of an aircraft, it is essential to define a suitable coordinate system.

- Inertial coordinate system: fixed to the earth and is used for aircraft motion analysis, with respect to earth.



(i) Earth-fixed coordinate—unchanged by altitude changes in the aircraft

- Body coordinate system is fixed to the moving aircraft.



(b meaning body fixed)

(ii) Body-fixed coordinate—translates and rotates with the aircraft

Forces and Moments

- The aircraft experiences

1. Longitudinal force
2. Lateral force and
3. Vertical (normal) force.

- Likewise, there are three moments:

1. Rolling moment about longitudinal axis
2.Pitching moment about lateral axis
3. Yawing Moment about vertical axis

G N C

- ***Guidance*** refers to the determination of the desired path of travel (the "trajectory") from the vehicle's current location to a designated target, as well as desired changes in velocity, rotation and acceleration for following that path.
- ***Navigation*** refers to the determination, at a given time, of the vehicle's location and velocity (the "state vector") as well as its attitude.
- ***Control*** refers to the manipulation of the forces, by way of steering controls, thrusters, etc., needed to execute guidance commands whilst maintaining vehicle stability

What is an Aircraft Control System?

- A control system is a collection of mechanical and electronic equipment that allows an aircraft to be flown with exceptional precision and reliability.

- A control system consists of cockpit controls, sensors, actuators (hydraulic, mechanical or electrical) and computers.

Aircraft engine controls are also considered as flight controls as they change speed.

Aircraft flight control systems can be divided into two main groups:

- Primary flight control
- Secondary flight control

Primary flight controls are required to safely control an aircraft during flight and consist of ailerons, elevators and rudder.

Secondary flight controls are intended to improve the aircraft performance characteristics or to relieve excessive control loading, and consist of high lift devices such as slats and flaps etc

Types of Flight Control Systems

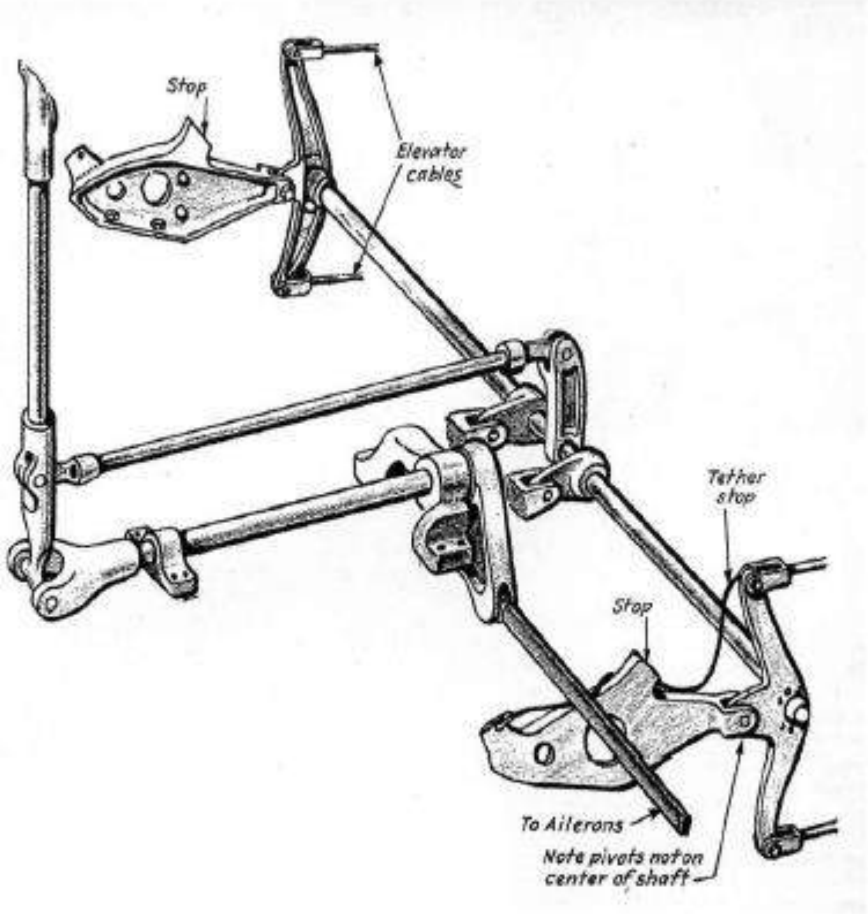
- Mechanical
- Hydraulic-Mechanical
- Fly-by-wire

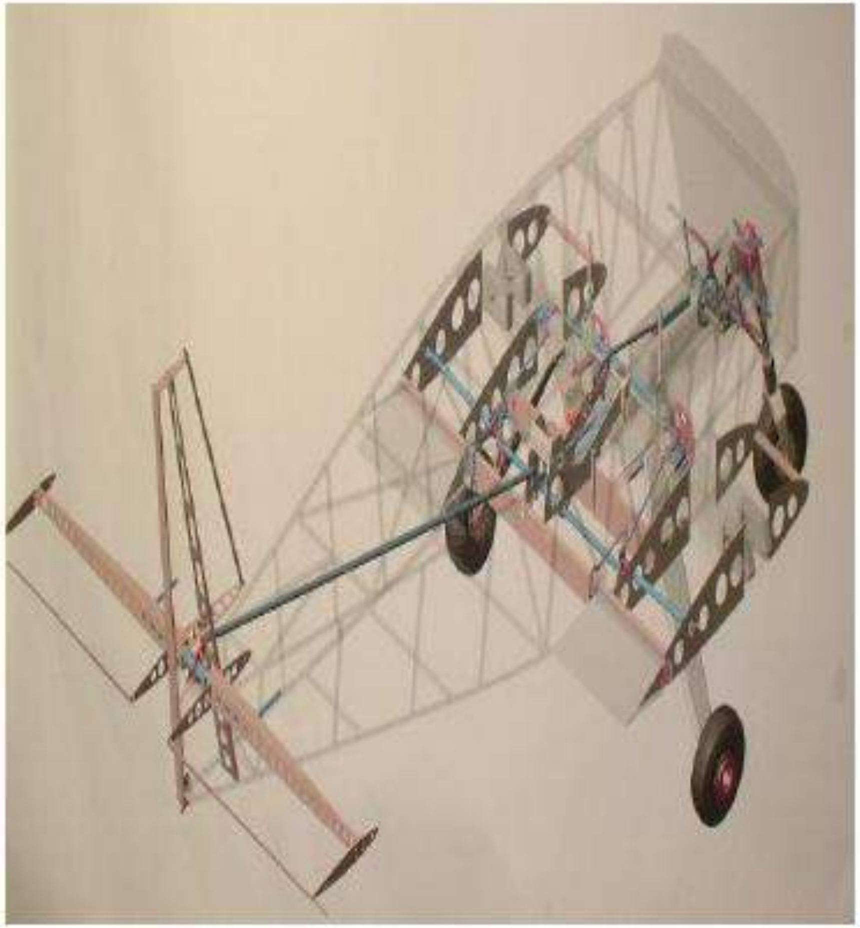
Mechanical Flight Control System

- Basic method of controlling an aircraft
- Used in early aircraft and currently in small aircraft where the aerodynamic forces are not excessive.
- It uses a collection of mechanical parts such as rods, tension cables, pulleys, counterweights, and sometimes chains to transmit the forces applied from the cockpit controls directly to the control surfaces

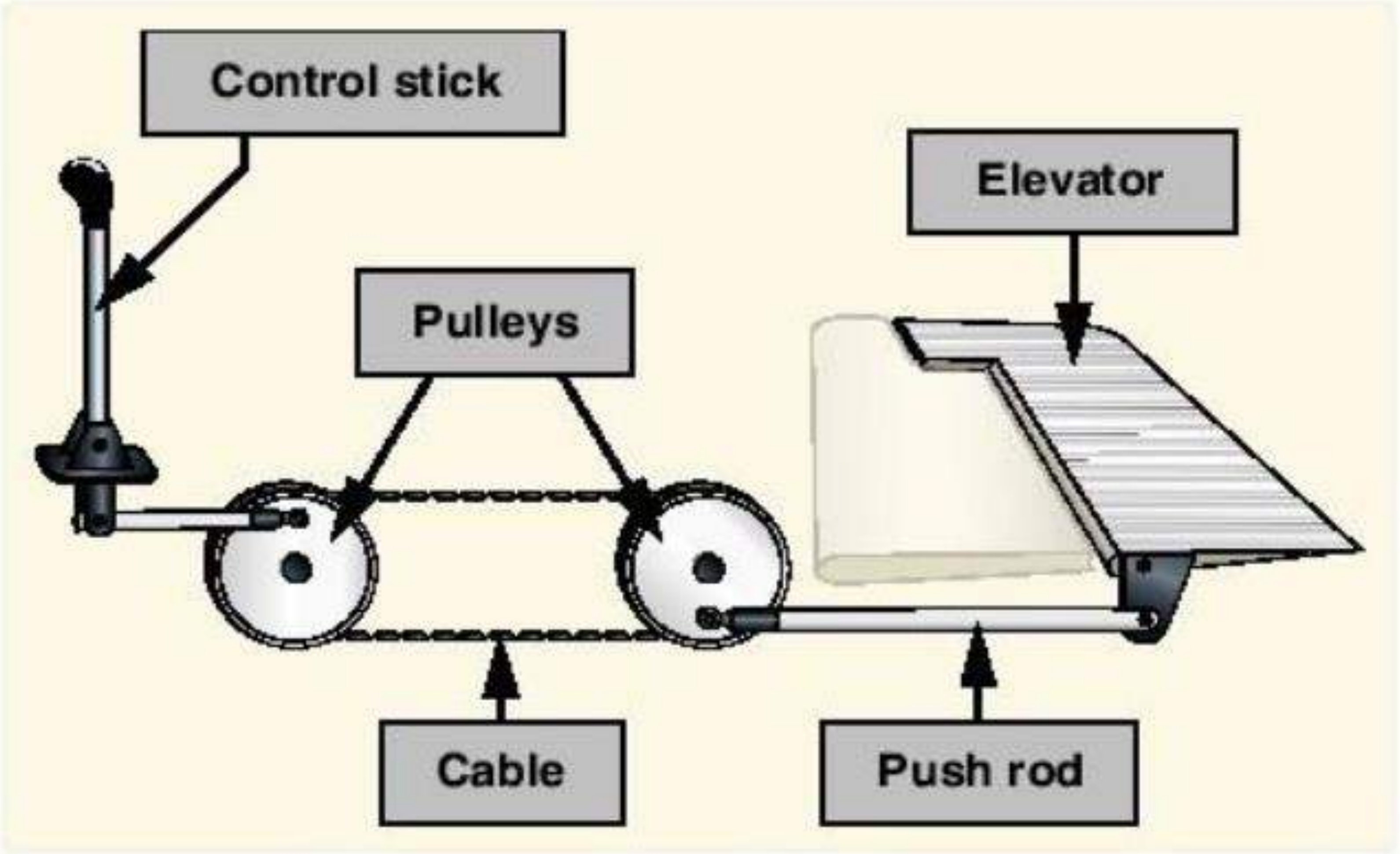
Conventional Flight Control System Components

Push Pull Rods

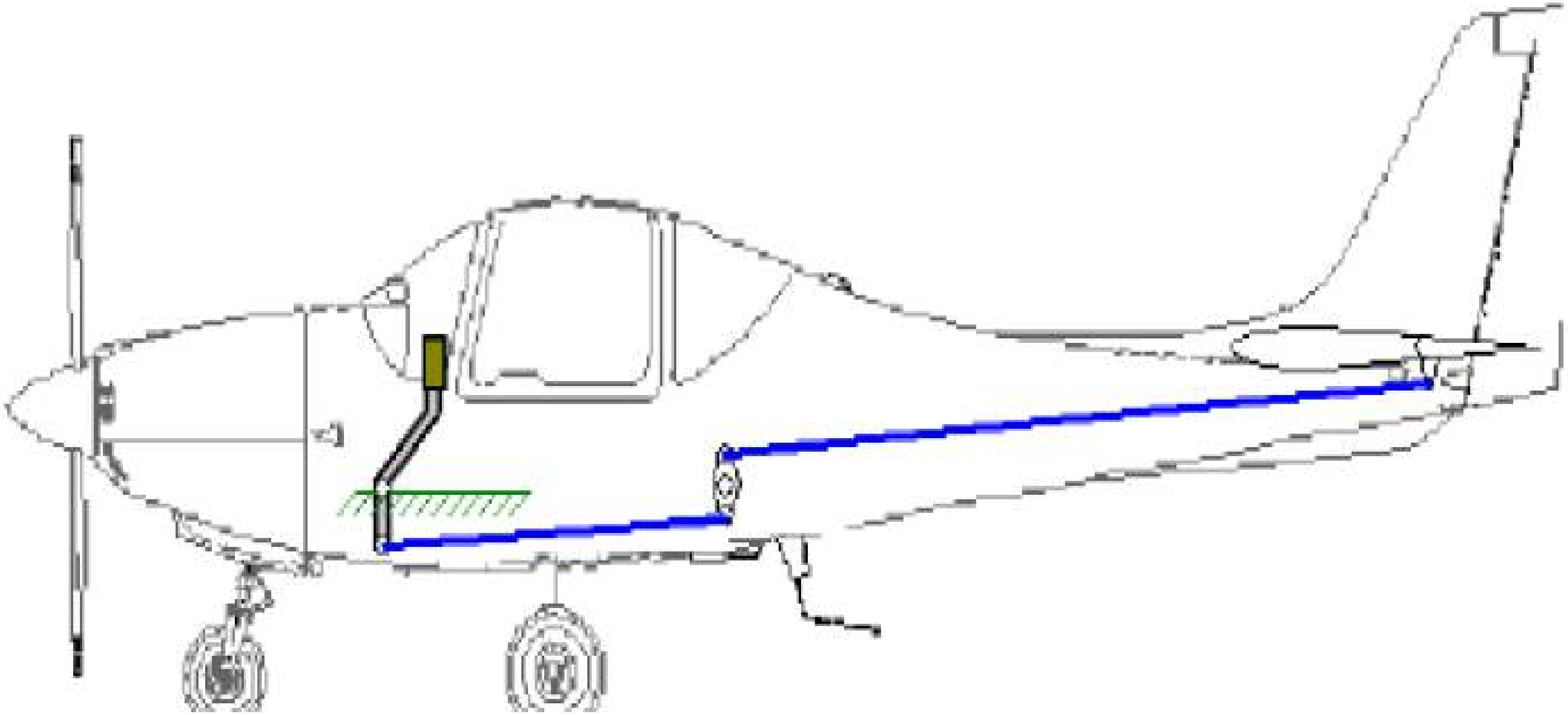




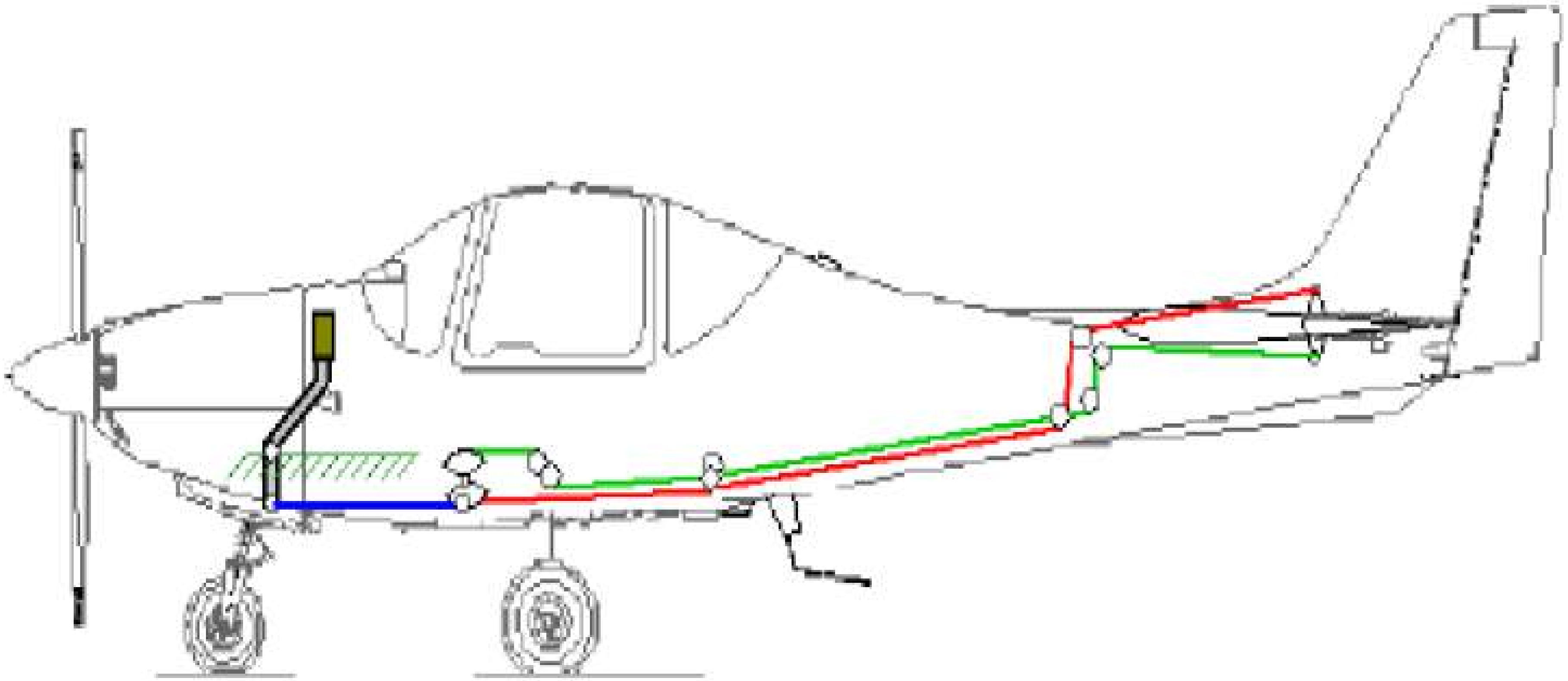
Mechanical Flight Control System



Push Pull Rod System for Elevator Control



Cables & Pulleys System for Elevator Control

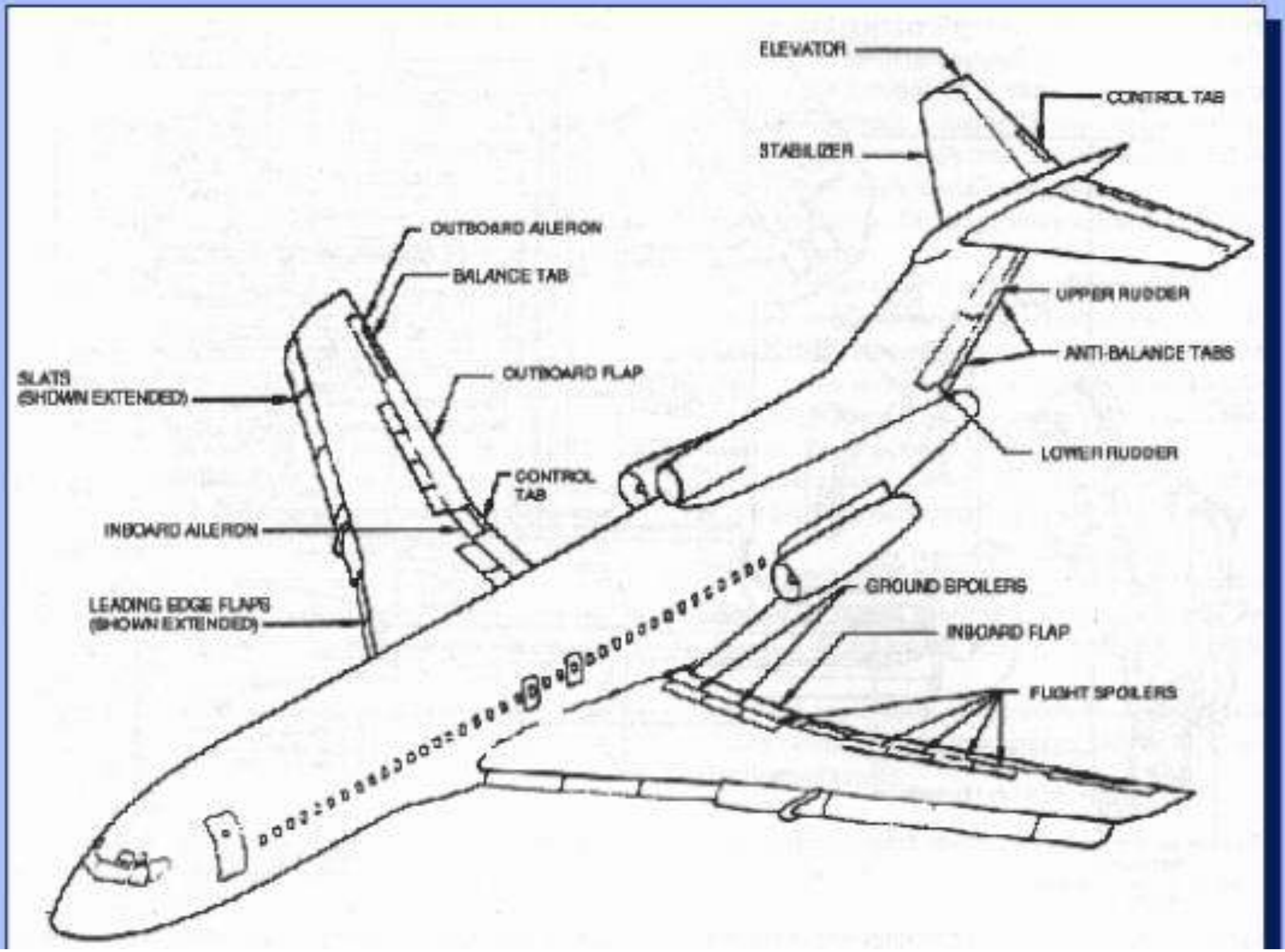


- Increases in the control surface area required by large aircraft or higher loads caused by high airspeeds in small aircraft lead to a large increase in the forces needed to move them, consequently complicated mechanical gearing arrangements were developed to extract maximum mechanical advantage in order to reduce the forces required from the pilots. This arrangement can be found on bigger or higher performance propeller aircraft such as the Fokker 50.

- Some mechanical flight control systems use **Servo tabs** that provide aerodynamic assistance. Servo tabs are small surfaces hinged to the control surfaces. The flight control mechanisms move these tabs, aerodynamic forces in turn move, or assist the movement of the control surfaces reducing the amount of mechanical forces needed. This arrangement was used in early piston-engined transport aircraft and in early jet transports. The Boeing 737 incorporates a system, whereby in the unlikely event of total hydraulic system failure, it automatically and seamlessly reverts to being controlled via servo-tab.

Modern Advanced Aircraft Have Many Control Surfaces

- Each set of control surfaces has a different purpose
- The pilot cannot control each surface directly - there are just too many!
- A flight control system is used to tell which control surfaces to move, and by how much, based on simple inputs from the pilot.



Need for Powered Control System

- The Complexity and Weight of the system (Mechanical) increased with Size and Performance of the aircraft.
- When the pilot's action is not directly sufficient for the control, the main option is a powered system that assists the pilot.
- The hydraulic system has demonstrated to be a more suitable solution for actuation in terms of reliability, safety, weight per unit power and flexibility, with respect to the electrical system

Hydraulic Mechanical Flight Controls

- Sometimes referred to as “boosted”
- Adds complexity to an already complex system
  - Added weight
- Has to include hydraulic lines, actuators, pumps, and a linkage between the hydraulic system and the mechanical cockpit controls
- Makes flying the aircraft less demanding and allows for high loads on control surfaces and much physically larger control surfaces

Powered Assisted Control System

- The pilot, via the cabin components, sends a signal, or demand, to a valve that opens ports through which high pressure hydraulic fluid flows and operates one or more actuators.
- The valve, that is located near the actuators, can be signalled in two different ways: mechanically or electrically
- Mechanical signalling is obtained by push-pull rods, or more commonly by cables and pulleys
- Electrical signalling is a solution of more modern and sophisticated vehicles

Disadvantages of Mechanical and Hydro-Mechanical Systems

- Heavy and require careful routing of flight control cables through the aircraft using pulleys, cranks, tension cables and hydraulic pipes.
- They require redundant backup to deal with failures, which again increases weight.
- Limited ability to compensate for changing aerodynamic conditions

Fly –By –Wire System (FBW)

- The term "fly-by-wire" implies a purely electrically-signalled control system
- It is a computer-configured controls, where a computer system is interposed between the operator and the final control actuators or surfaces
- It modifies the manual inputs of the pilot in accordance with control parameters
- These are carefully developed and validated in order to produce maximum operational effect without compromising safety

- The FBW architecture was developed in 1970's
- Initially starting as an analogue technique and later on transformed into digital.
- It was first developed for military aviation, where it is now a common solution
- The supersonic Concorde can be considered a first and isolated civil aircraft equipped with a (analogue) fly-by-wire system

Operation

- The pilot’s demand is first of all transduced into electrical signal in the cabin and sent to a group of independent computers (Airbus architecture substitute the cabin control column with a side stick)

- The computers sample also data concerning the flight conditions and servo-valves and actuators positions

- The pilot’s demand is then processed and sent to the actuator, properly tailored to the actual flight status.

- The flight data used by the system mainly depend on the aircraft category; in general the following data are sampled and processed:
  - pitch, roll, yaw rate and linear accelerations
  - Angle of attack and sideslip
  - Airspeed/Mach number, Pressure, Altitude and radio altimeter indications
  - Stick and pedal demands
  - Other cabin commands such as landing gear condition, thrust lever position, etc.

- The full system has high redundancy to restore the level of reliability of a mechanical or hydraulic system, in the form of multiple (triplex or quadruplex) parallel and independent lanes to generate and transmit the signals, and independent computers that process them

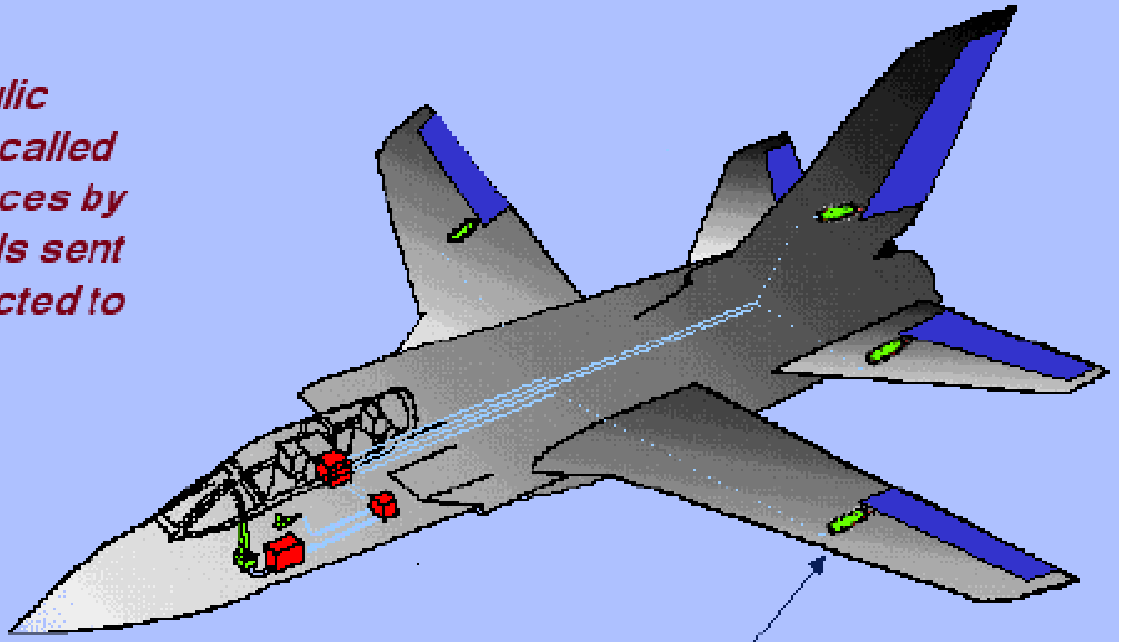
Other Flight Controls

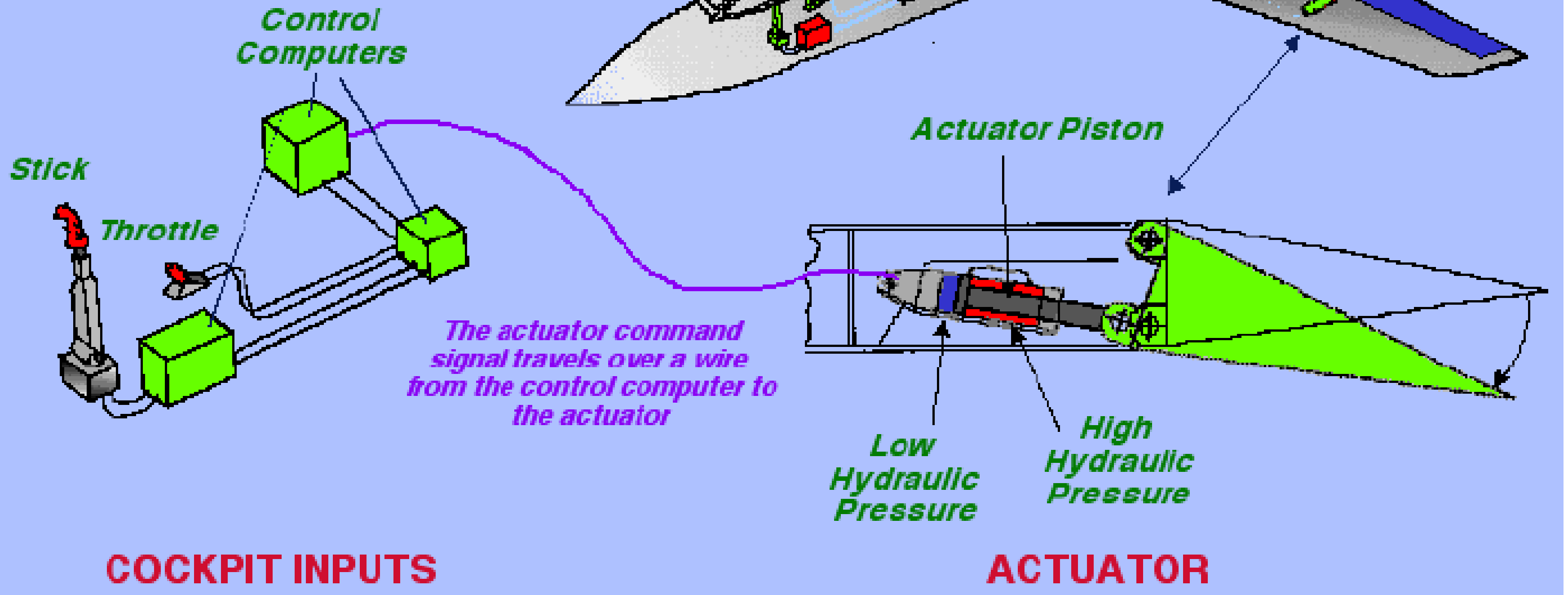
- Electrically actuated flight controls which uses electric motors to move control surfaces.
- Closed circuit hydraulic controls, each individual control surface has its own full hydraulic system, eliminates a central system
- Intelligent flight controls, uses neural networks to determine changes in flight dynamics and adjust control laws accordingly


"Putting it All Together"
Flight Control System
Aircraft Sensors
Position Gyros
Rate Gyros
etc.
• Orientation
• Velocity
• Altitude
• etc.
Sensor
Measurements
Flight
Control
Computer
Pilot
Commands
Aircraft Cockpit
• Flight Path Command
• Velocity Command
• Altitude Command
• etc.
Controller Commands
• Throttle Position
• Rudder Position
• Elevator Position
• Aileron Position
• etc.
Ailerons
Rudder
Engines
Elevator
Aircraft ControEffectors

Control Surfaces are Moved with Actuators

In modern aircraft, hydraulic systems or electric motors called <u>actuators</u>move control surfaces by responding to control signals sent from a flight computer connected to the control stick.

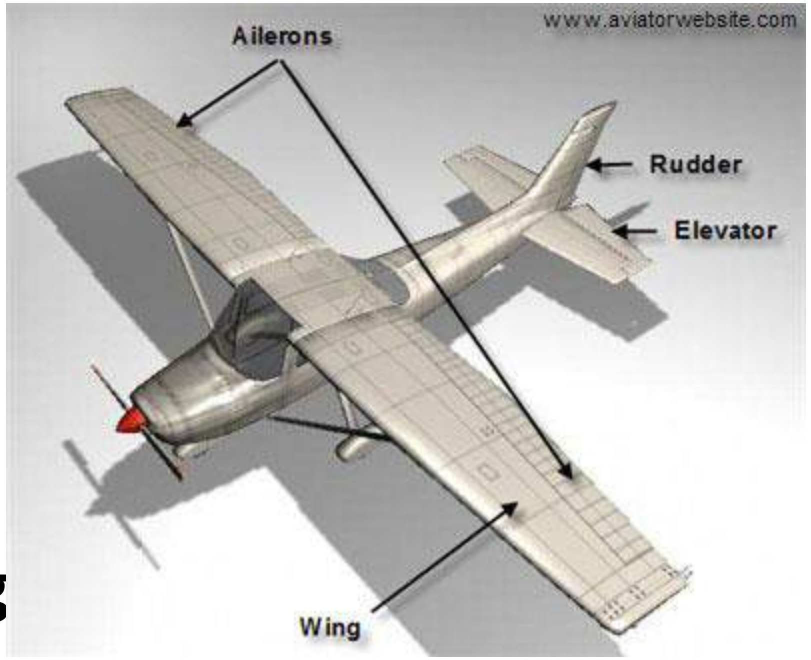




Flight Control Systems
Flight Controls

- Aircraft flight control systems consist of primary and secondary systems.

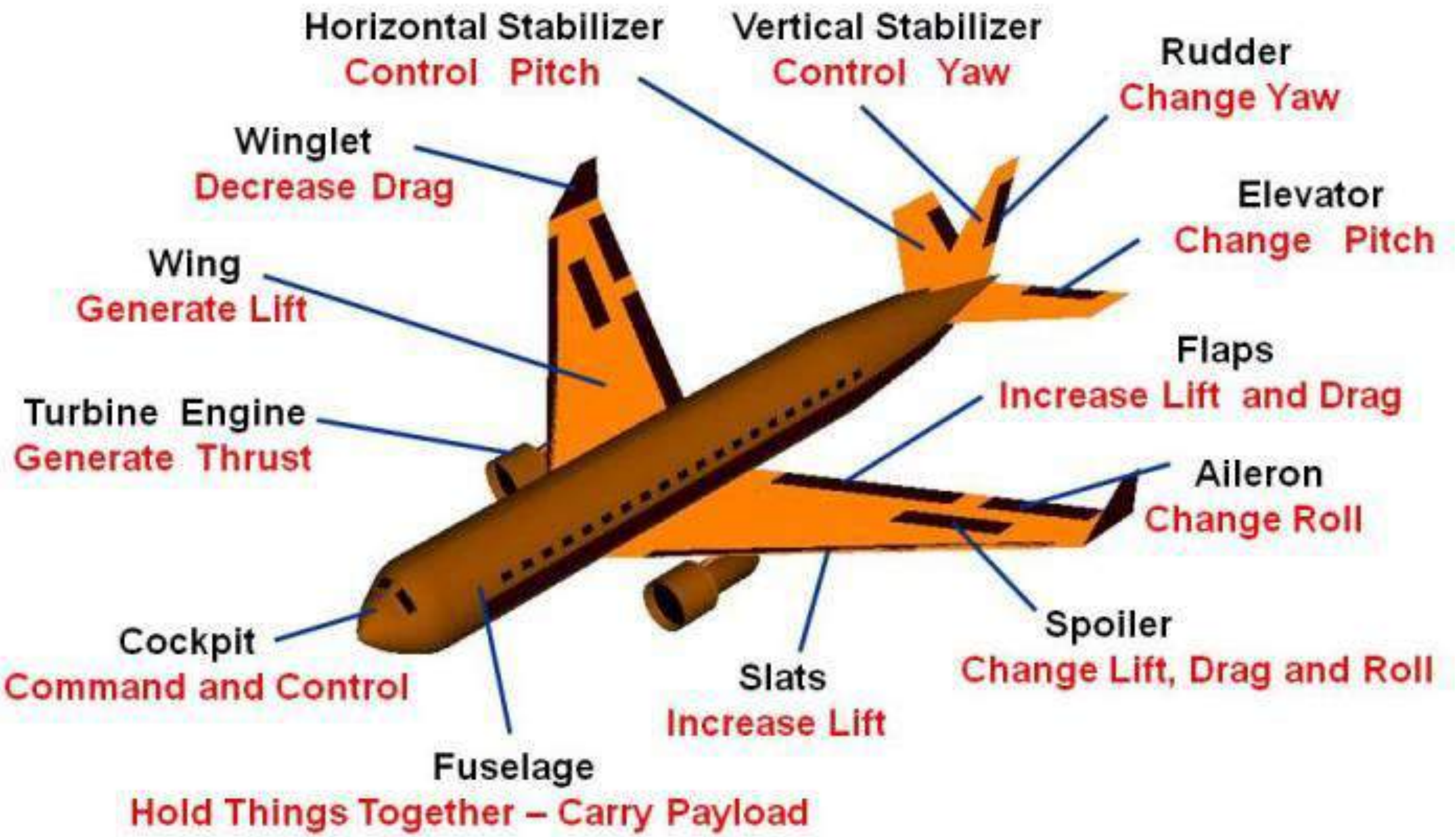
- The ailerons, elevator (or stabilator), and rudder constitute the primary control system and are required to control an aircraft safely during flight.

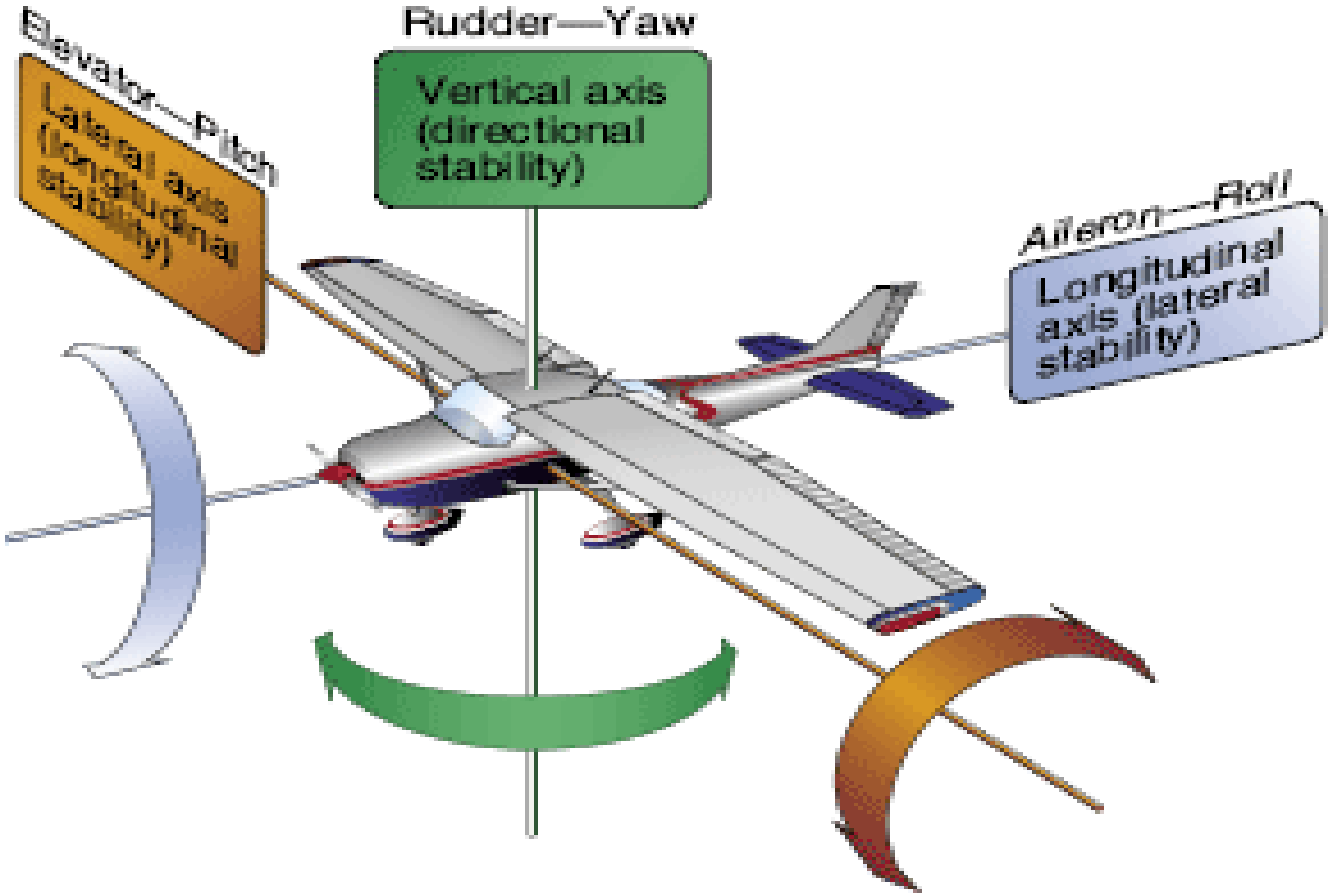


- **Wing flaps, leading edge devices, spoilers etc constitute the secondary control system**

- **They improve the performance characteristics of the airplane or relieve the pilot of excessive control forces.**

Airplane Parts and Function

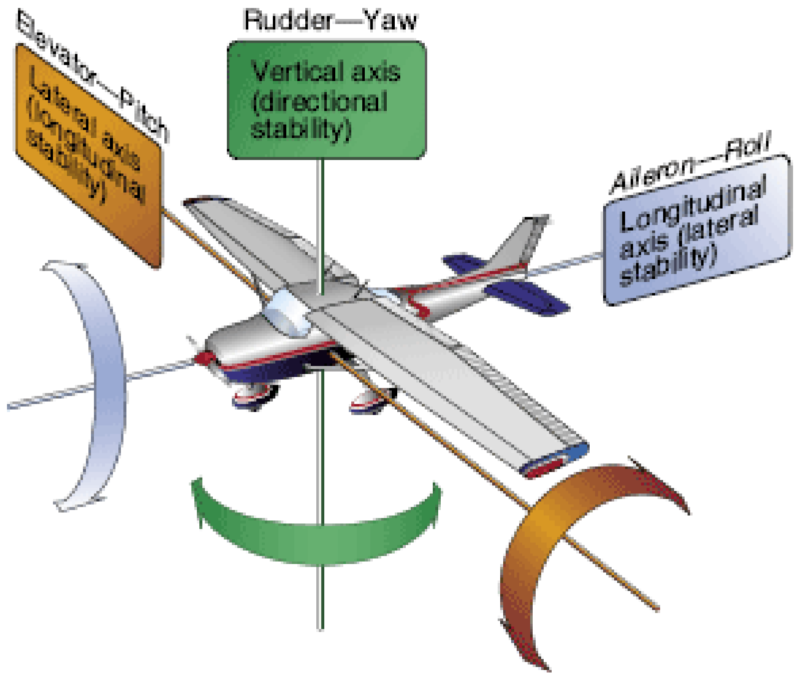




Primary Control Surface	Airplane Movement	Axes of Rotation	Type of Stability
Aileron	Roll	Longitudinal	Lateral
Elevator/ Stabilator	Pitch	Lateral	Longitudinal
Rudder	Yaw	Vertical	Directional

Flight Control Systems
Primary Flight Controls

- A properly designed airplane is stable and easily controlled during normal maneuvering.
- Control surface inputs cause movement about the three axes of rotation.



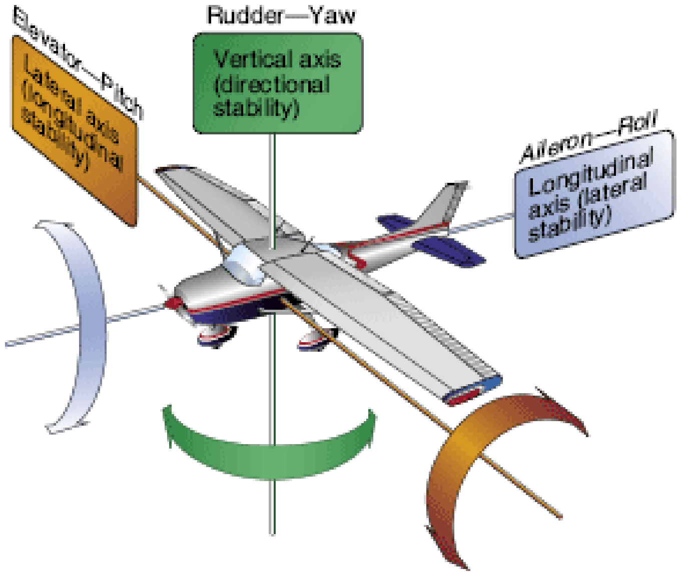
Primary Control Surface	Airplane Movement	Axes of Rotation	Type of Stability
Aileron	Roll	Longitudinal	Lateral
Elevator/ Stabilator	Pitch	Lateral	Longitudinal
Rudder	Yaw	Vertical	Directional

Flight Control Systems
Ailerons

- Ailerons control roll about the longitudinal axis.

- The ailerons are attached to the outboard trailing edge of each wing and move in the opposite direction from each other.

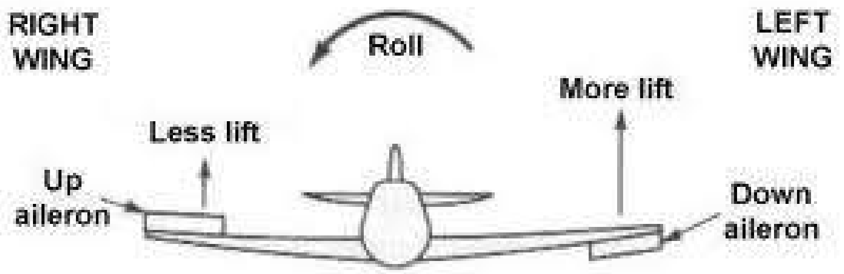
- Ailerons are connected by cables, bellcranks, pulleys and/or push-pull tubes to a control wheel or control stick.

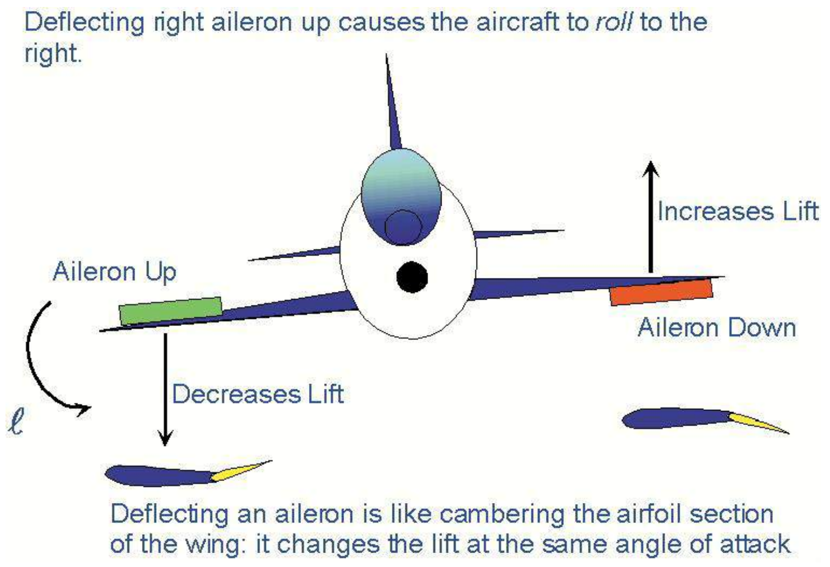


Primary Control Surface	Airplane Movement	Axes of Rotation	Type of Stability
Aileron	Roll	Longitudinal	Lateral
Elevator/ Stabilator	Pitch	Lateral	Longitudinal
Rudder	Yaw	Vertical	Directional

- Moving the control wheel or control stick to the right causes the right aileron to deflect upward and the left aileron to deflect downward.

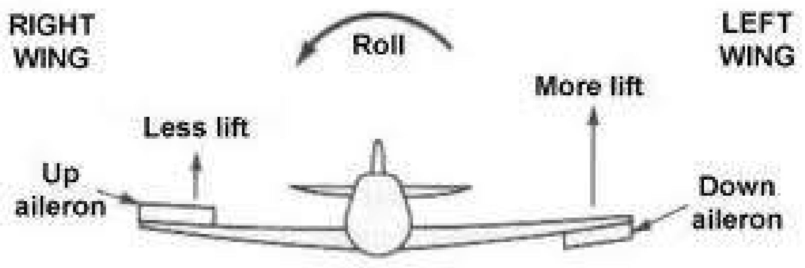
- The upward deflection of the right aileron decreases the camber resulting in decreased lift on the right wing.



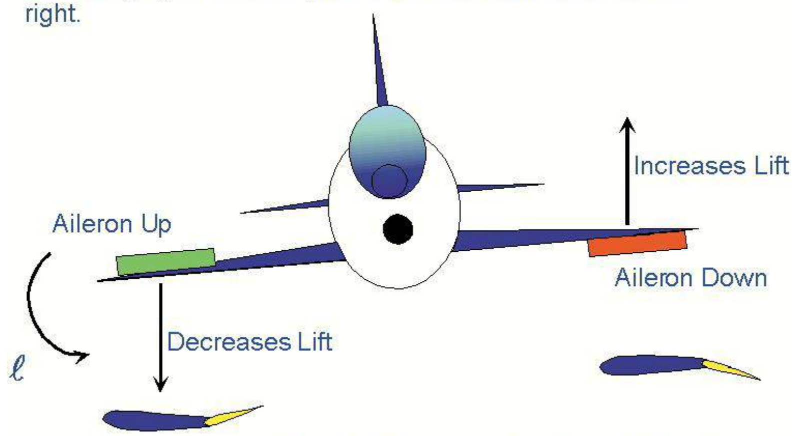


- The corresponding downward deflection of the left aileron increases the camber resulting in increased lift on the left wing.

- Thus, the increased lift on the left wing and the decreased lift on the right wing causes the airplane to roll to the right.



Deflecting right aileron up causes the aircraft to *roll* to the right.

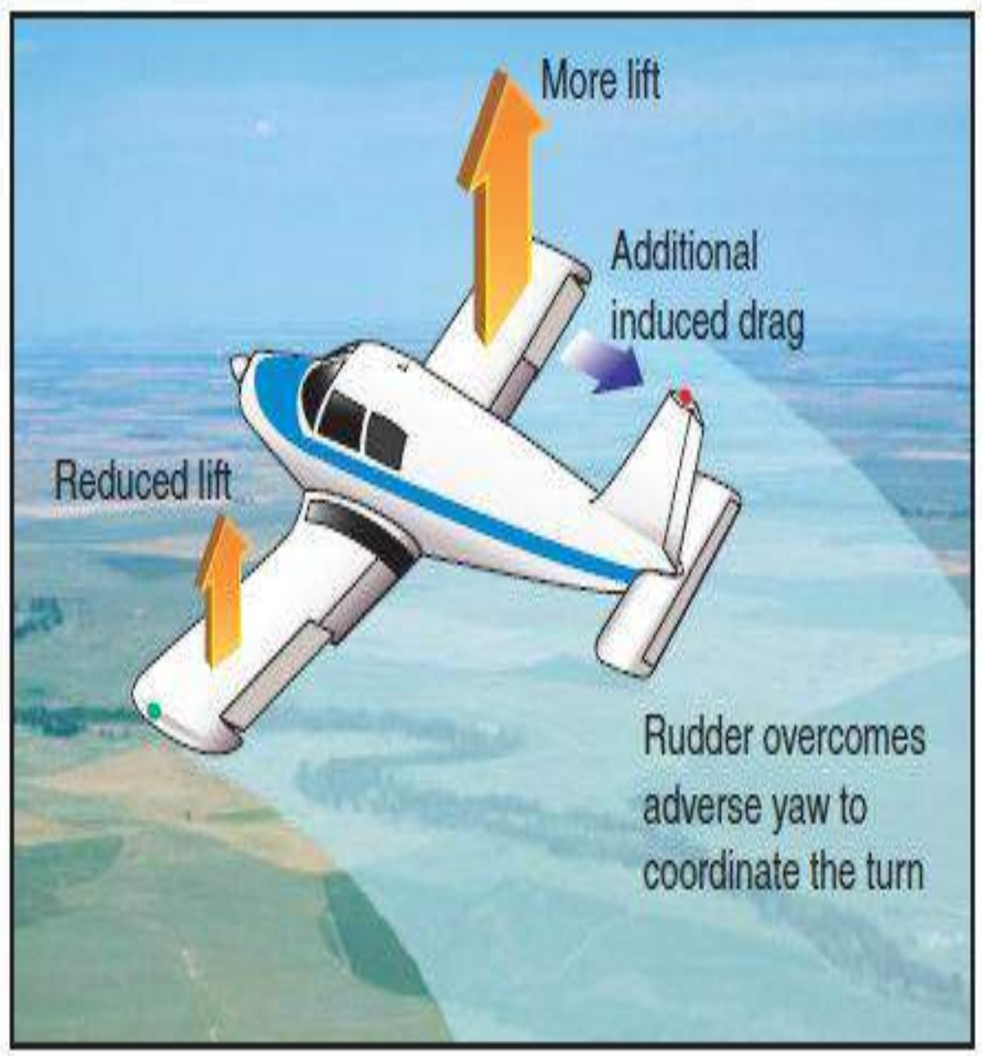


Deflecting an aileron is like cambering the airfoil section of the wing: it changes the lift at the same angle of attack

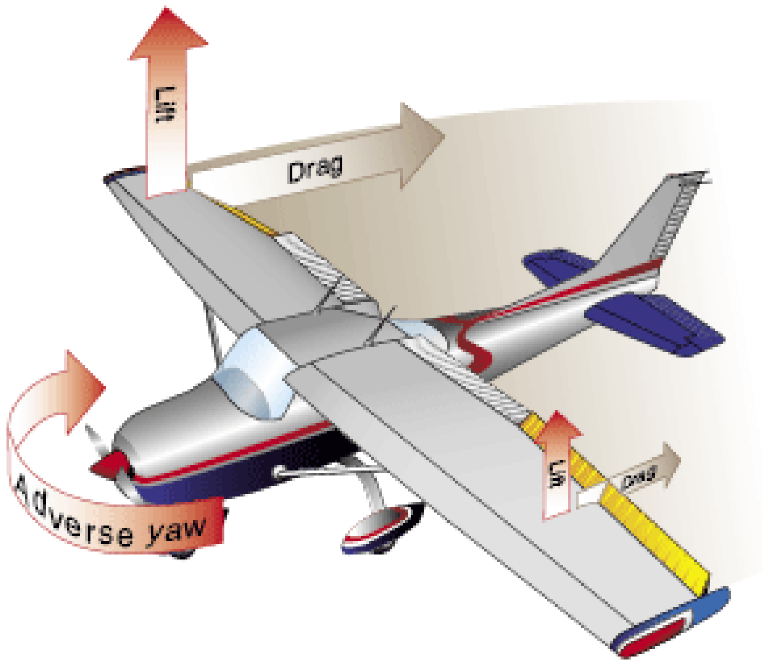
Flight Control Systems
Adverse Yaw

- **Since the downward deflected aileron produces more lift as evidenced by the wing raising, it also produces more drag.**

- **This added drag causes the wing to slow down slightly.**

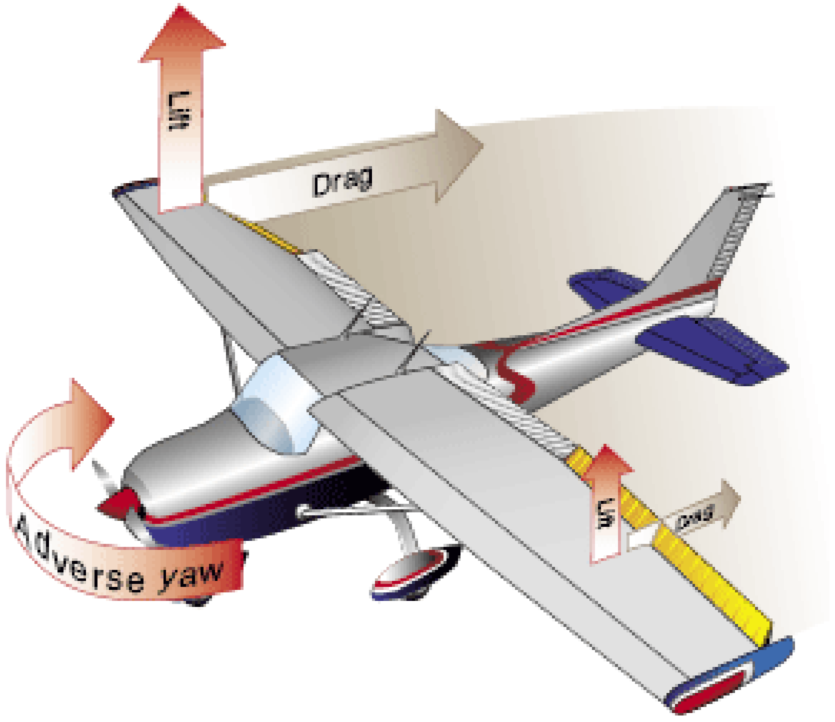


- **This results in the aircraft yawing toward the wing which had experienced an increase in lift (and drag).**



- Adverse yaw becomes more pronounced at low airspeeds.

- **At these slower airspeeds aerodynamic pressure on control surfaces are low and larger control inputs are required to effectively maneuver the airplane.**



- **Application of rudder is used to counteract adverse yaw.**

- **The amount of rudder control required is greatest at low airspeeds, high angles of attack, and with large aileron deflections.**

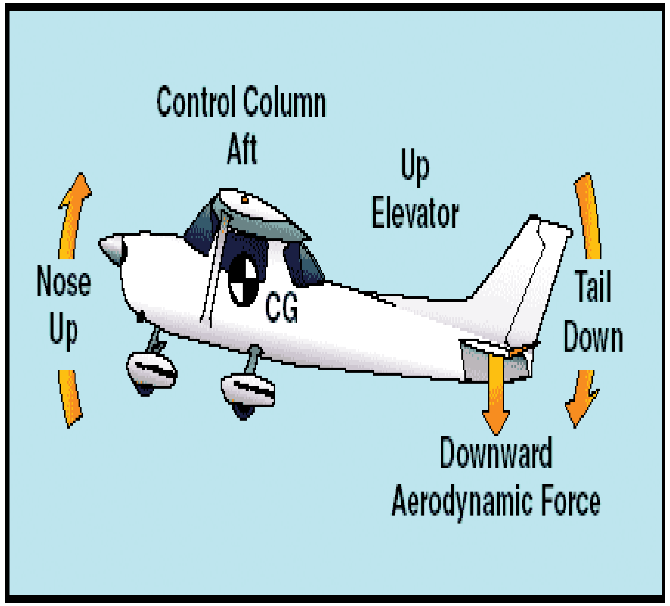
- **All turns are coordinated by use of ailerons, rudder, and elevator.**

- **Applying aileron pressure is necessary to place the aircraft in the desired angle of bank, while simultaneous application of rudder pressure is necessary to counteract the resultant adverse yaw.**

Flight Control Systems
Elevator

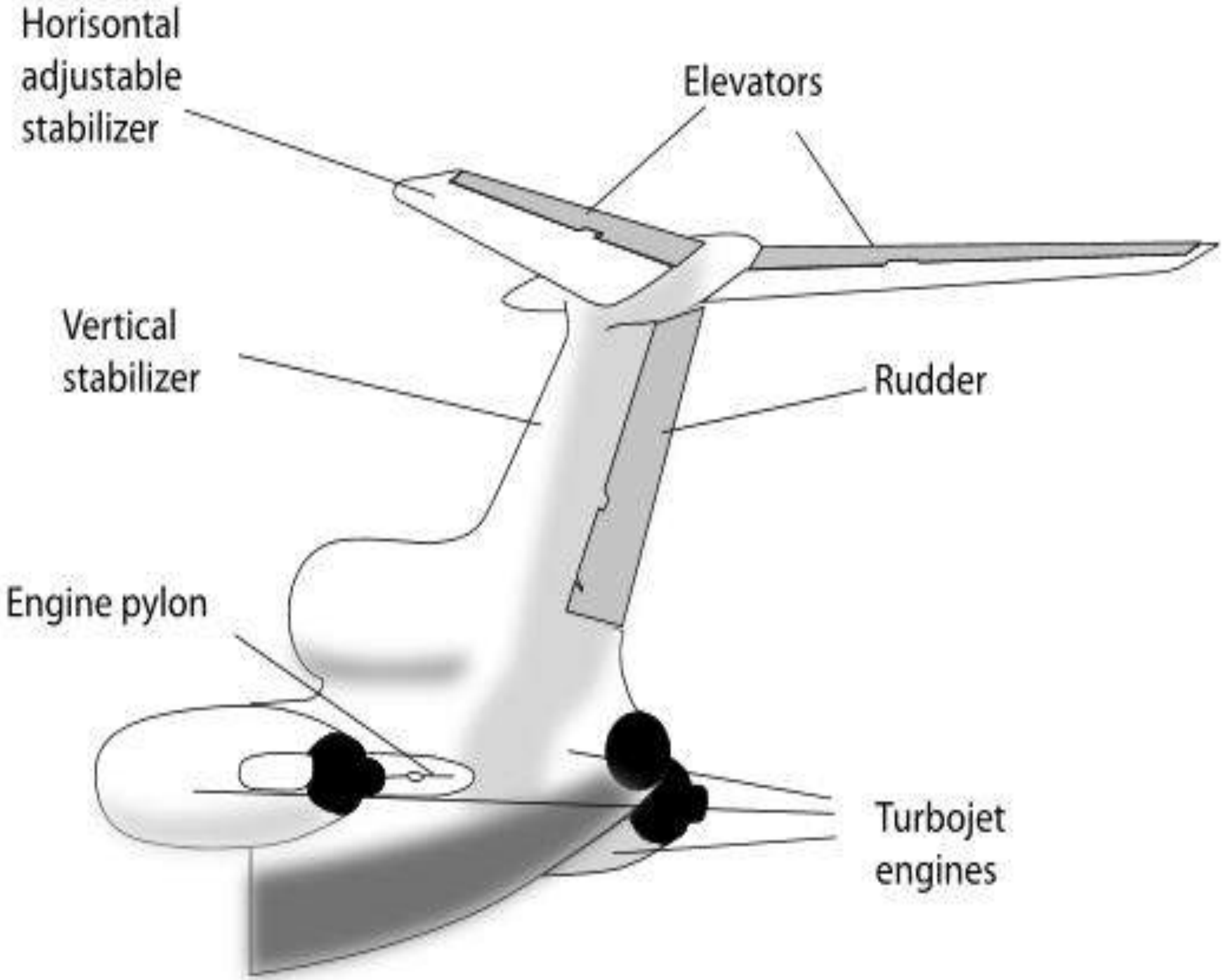
- The up-elevator position decreases the camber of the elevator and creates a downward aerodynamic force.

- The overall effect causes the tail of the aircraft to move down and the nose to pitch up.



Flight Control Systems
T-Tail

- **In a T-tail configuration, the elevator is above most of the effects of downwash from the propeller as well as airflow around the fuselage and/or wings during normal flight conditions.**



Horisontal adjustable stabilizer
Elevators
Vertical stabilizer
Rudder
Engine pylon
Turbojet engines

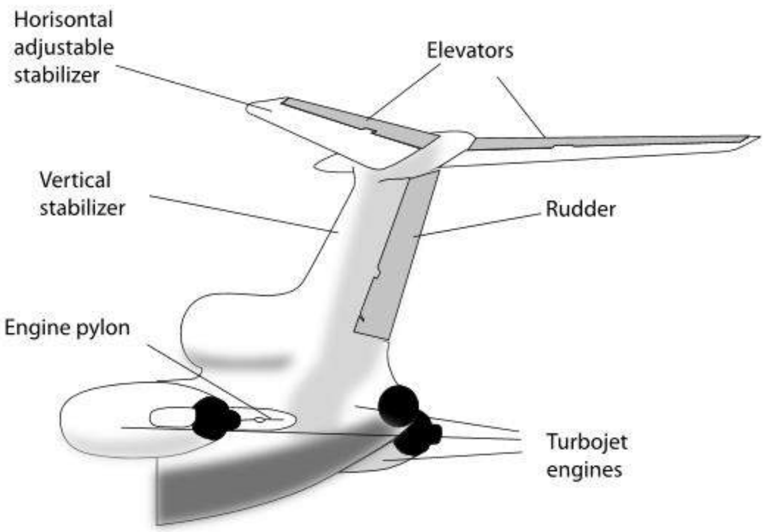
- **The horizontal tail surfaces may be attached near the lower part of the vertical stabilizer, at the midpoint, or at the high point, as in the T-tail design.**

- **Operation of the elevators in this undisturbed air allows control movements that are consistent throughout most flight regimes.**

- **T-tail designs have become popular on many light and large aircraft.**

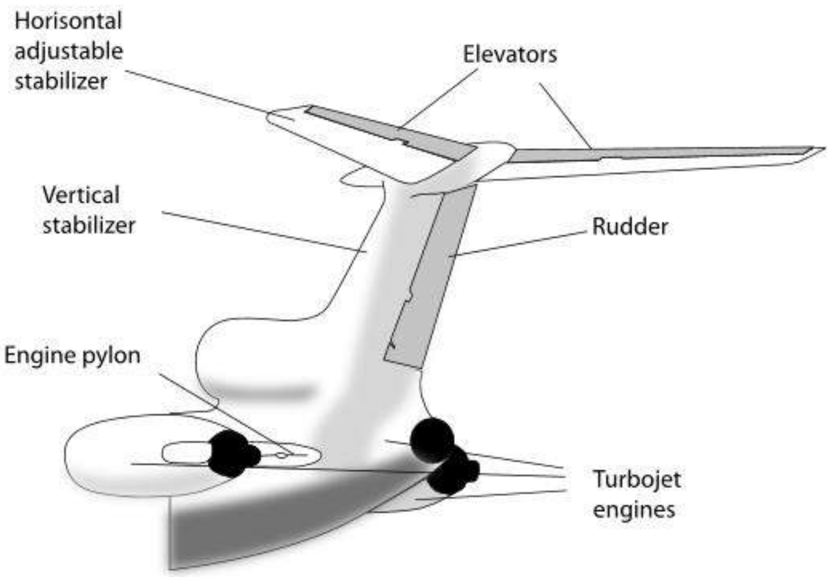
- T-tail configuration removes the tail from the exhaust blast of the engines.

- Seaplanes and amphibians often have T-tails in order to keep the horizontal surfaces as far from the water as possible.



- An additional benefit is reduced vibration and noise inside the aircraft.

- At slow speeds, the elevator on a T-tail aircraft must be moved through a larger number of degrees of travel to raise the nose a given amount than on a conventional-tail aircraft.



High Lift Devices

Spoilers

On low drag aircraft like sailplanes, spoilers are used to disrupt airflow over the wing and greatly increase the amount of drag. This allows a glider pilot to lose altitude without gaining excessive airspeed. Spoilers are sometimes called "lift dumpers". Spoilers that can be used asymmetrically are called spoilerons and are able to affect an aircraft's roll.

Slats

Slats, also known as Leading Edge Devices, are extensions to the front of a wing for lift augmentation, and are intended to reduce the stalling speed by altering the airflow over the wing. Slats may be fixed or retractable - fixed slats give excellent slow speed capabilities, but compromise higher speed performance. Retractable slats, as seen on most airliners, provide reduced stalling speed for take-off and landing, but are retracted for cruising.

DEFINITION OF STABILITY AND CONTROL

- There are two types of stability:

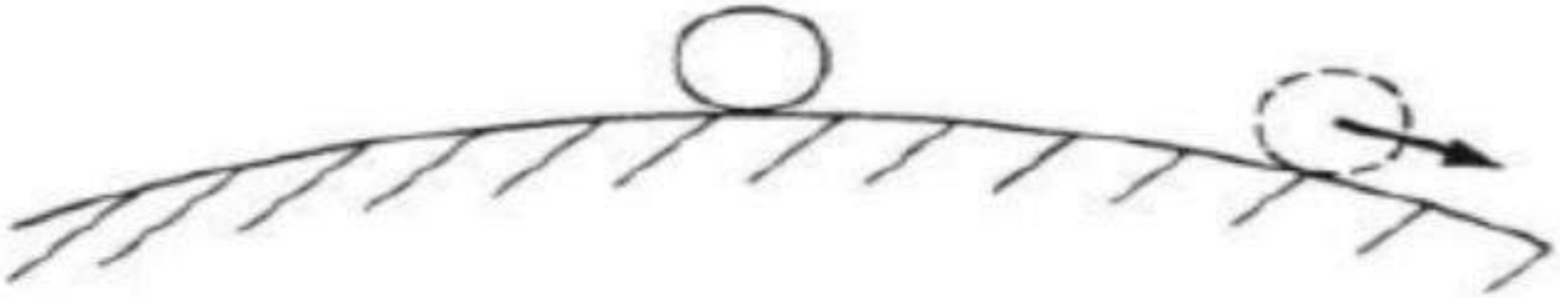
1) STATIC

2) DYNAMIC.

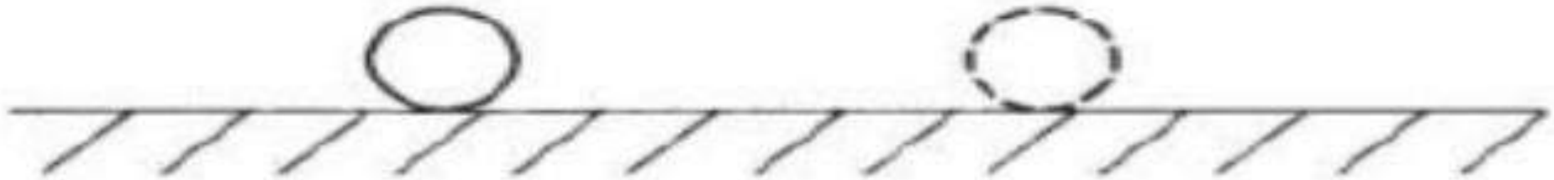
Static Stability

- If the forces and moments on the body caused by a disturbance tend initially to return the body toward its equilibrium position, the body is statically stable.
- The body has *positive* static stability.
- If the forces and moments are such that the body continues to move *away* from its equilibrium position after being disturbed, the body is *statically unstable.*
- The body has *negative* static stability
- Finally, imagine the marble on a flat horizontal surface as shown in Fig. c.
- Its moments are zero; it is in equilibrium. If the marble is disturbed to another location,it will still be in equilibrium. Such a system is *neutrally stable.*

(a)



(b)



Dynamic Stability

- Dynamic stability deals with the *time history* of the vehicle's motion after it initially responds to its static stability.
- Consider an airplane flying at an angle of attack α_e. such that its moments about the center of gravity are zero.
- The airplane is therefore in equilibrium at α_e called the *trim angle of attack.*

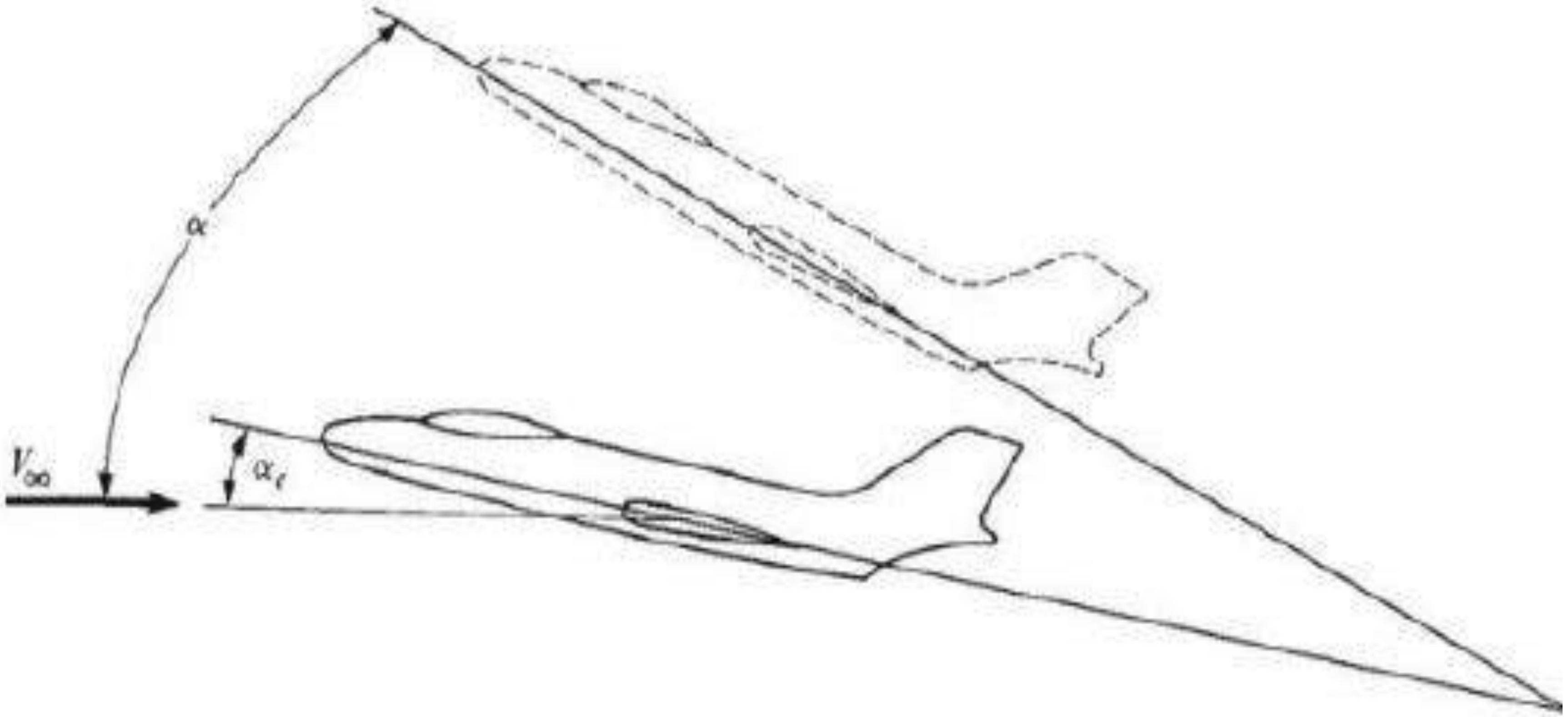
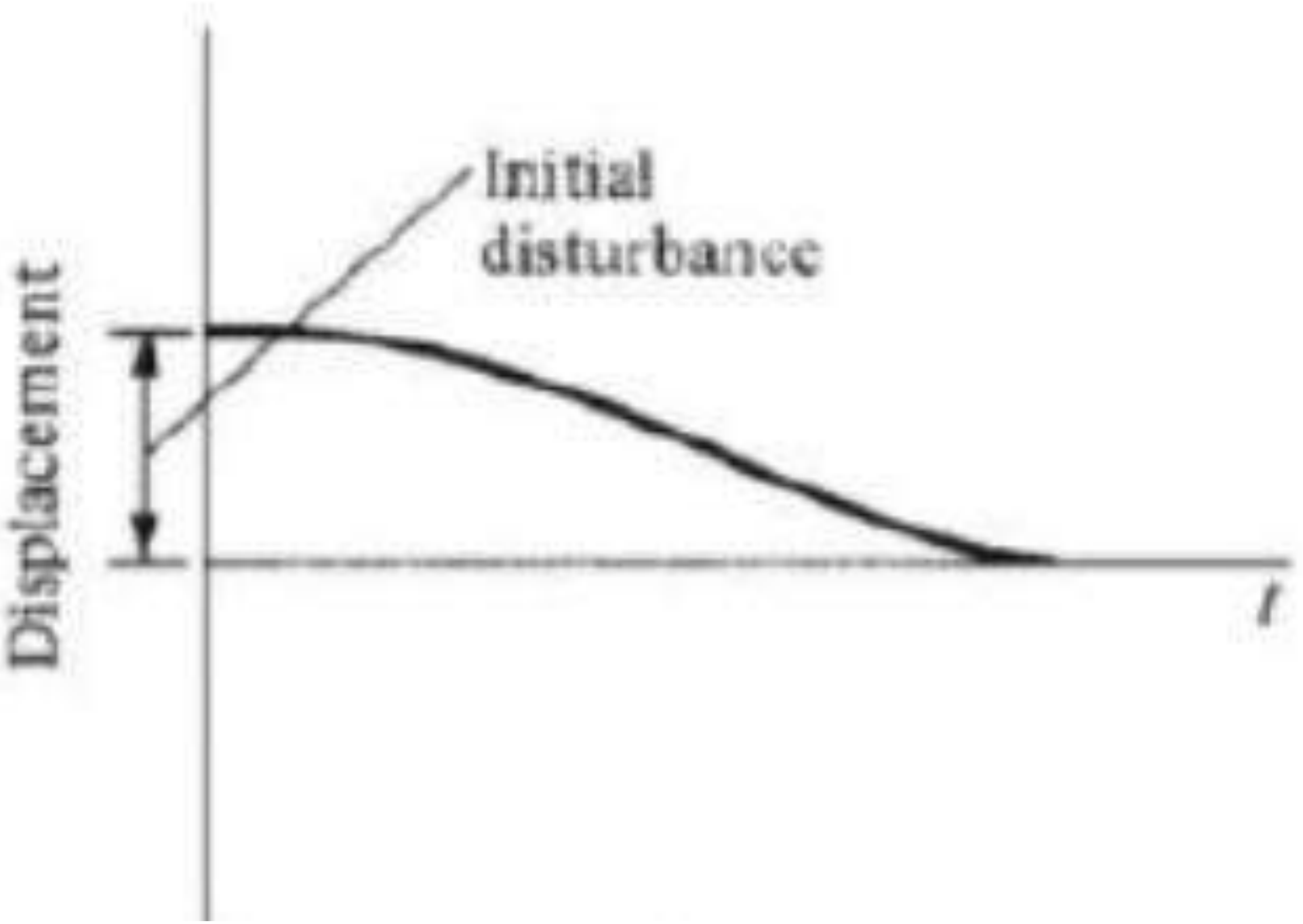
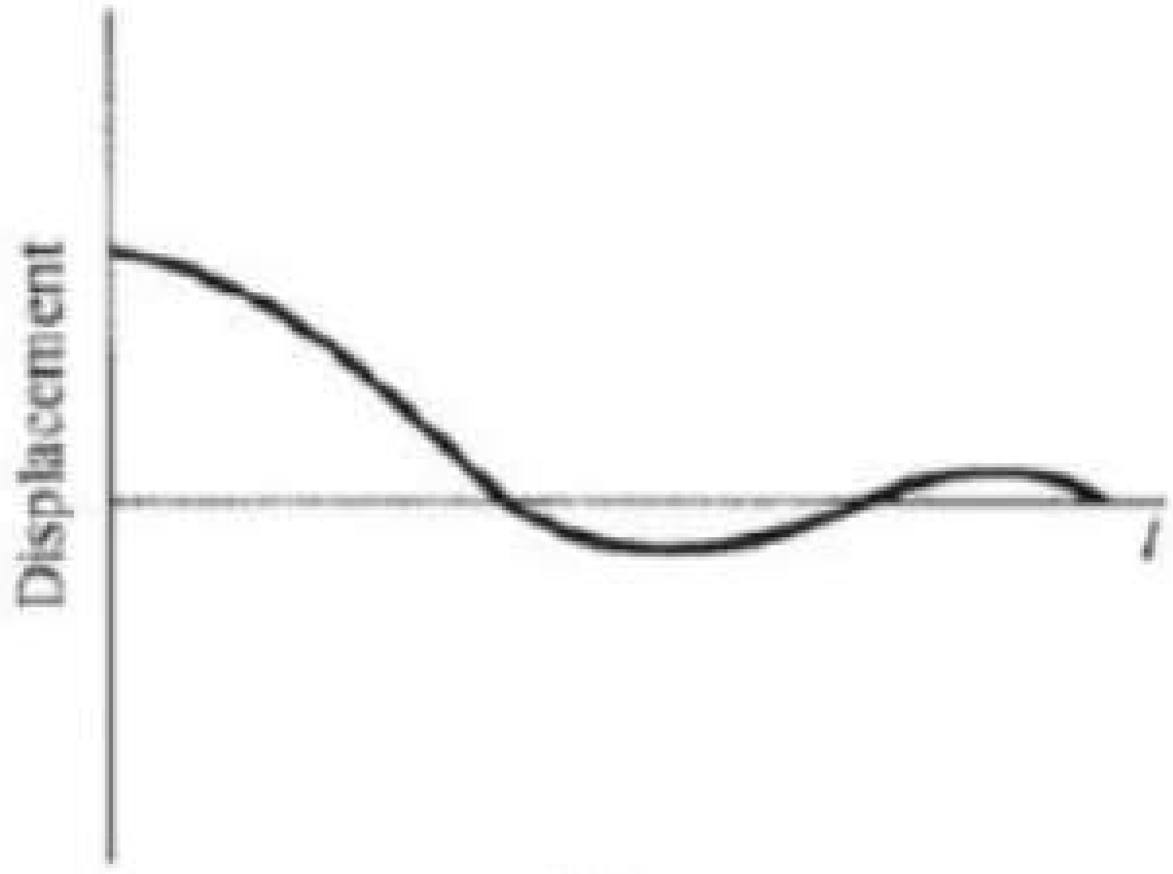


Figure 7.7 Disturbance from the equilibrium angle of attack.

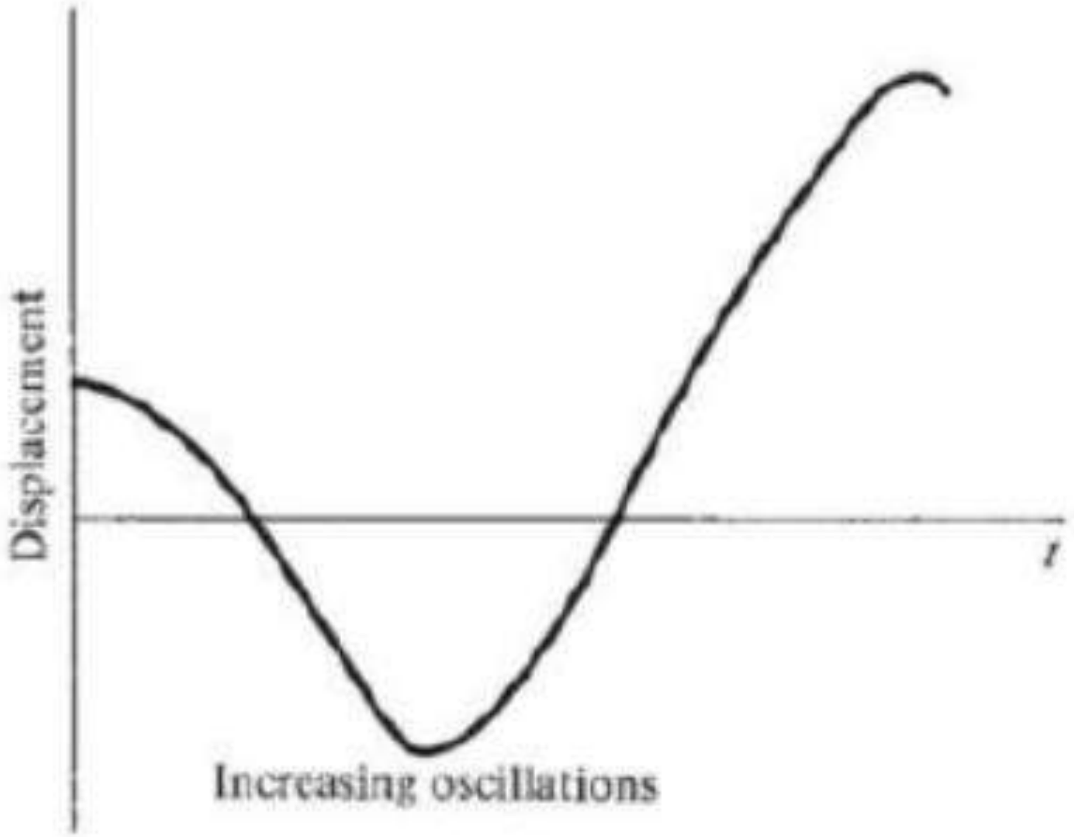
- Now assume that the airplane is disturbed to a new angle of attack α, as shown in Fig.
- The airplane has been pitched through a *displacement* $\alpha - \alpha_e$
- We can describe this motion by plotting the instantaneous displacement versus time, as shown in Fig. . Here $\alpha - \alpha_e$ is given as a function of *time t.*
- At $t = 0$ the displacement is equal to that produced by the gust.
- If the airplane is statically stable, it will *initially* tend to move back toward it equilibrium position; that is, $\alpha - \alpha_e$. will initially decrease.
- Over time the vehicle may "home in" to its equilibrium position, as shown in Fig. a. Such motion is called *aperiodic.*


Initial
disturbance
Displacement
t

- Alternatively, it may first overshoot the equilibrium position and approach αe after a series of oscillations with decreasing amplitude, as shown in Fig. b.
- Such motion is described as *damped oscillations*.
- In both situations, the airplane eventually returns to its equilibrium position after some interval of time.
- A body is dynamically stable if, of its own accord, it eventually returns to and remains at its equilibrium position over time.


Displacement
t

- In contrast, after initially responding to its static stability, the airplane may oscillate with increasing amplitude.
- Here the equilibrium position is never maintained for any period, and the airplane eventually diverges completely; the airplane in this case is *dynamically unstable.*
- Also, it is theoretically possible for the airplane to pitch back and forth with constant-amplitude oscillations.


Displacement
t
Increasing oscillations

Control

- The function of conventional control surfaces (elevators, ailerons, and rudder) on an airplane is usually

(1) to change the airplane from one equilibrium position to another and

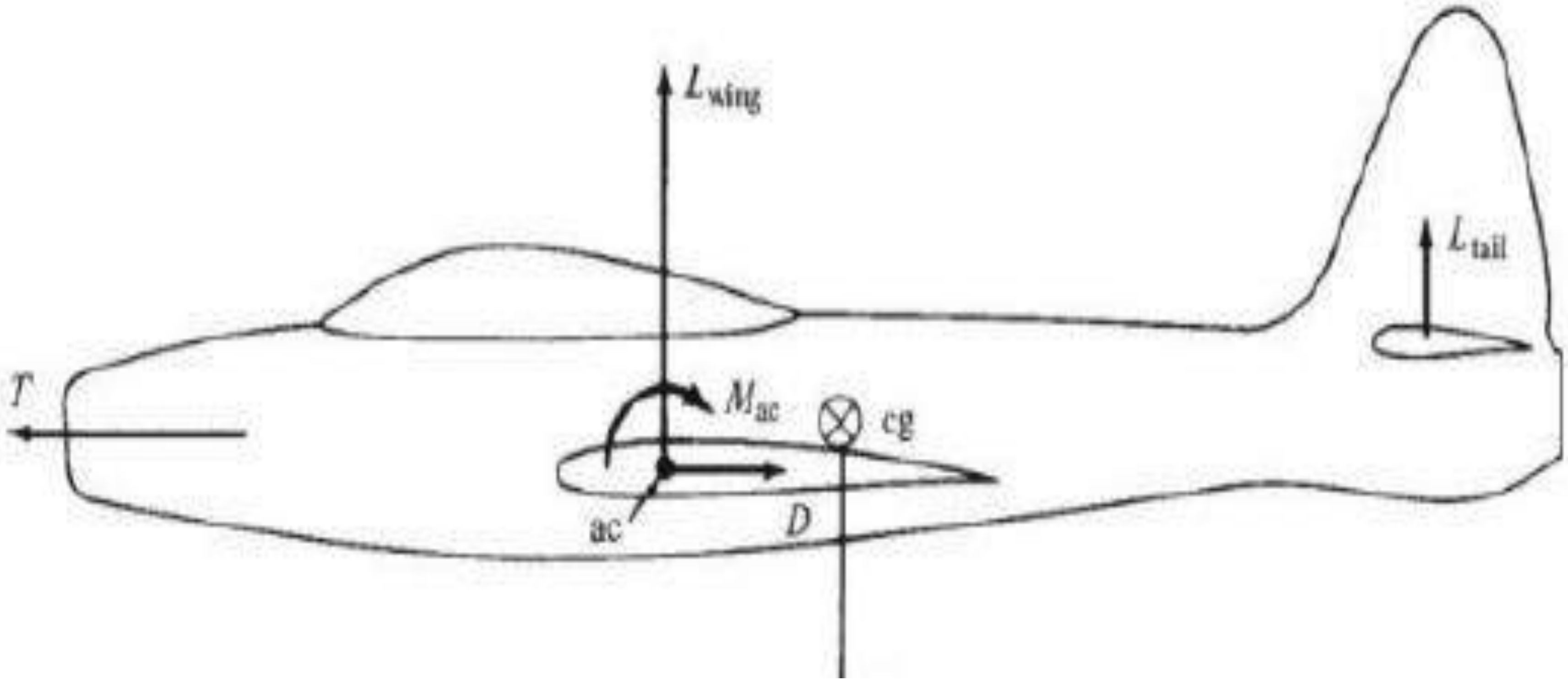
(2) to produce non-equilibrium accelerated motions such as maneuvers.

- The study of the *deflections* of the ailerons, elevators, and rudder necessary to make the airplane do what we want and of the amount of *force* that must be exerted by the pilot (or the hydraulic boost system) to deflect these controls is part of a discipline called *airplane control*

MOMENTS ON THE AIRPLANE

- A study of stability and control is focused on moments: moments on the airplane and moments on the control surfaces.
- There exists a particular point about which the moments are independent of the angle of attack. This point is defined as the *aerodynamic center* for the wing.
- The moment and its coefficient about the aerodynamic center are denoted by M_{ac} and $C_{M,ac}$.

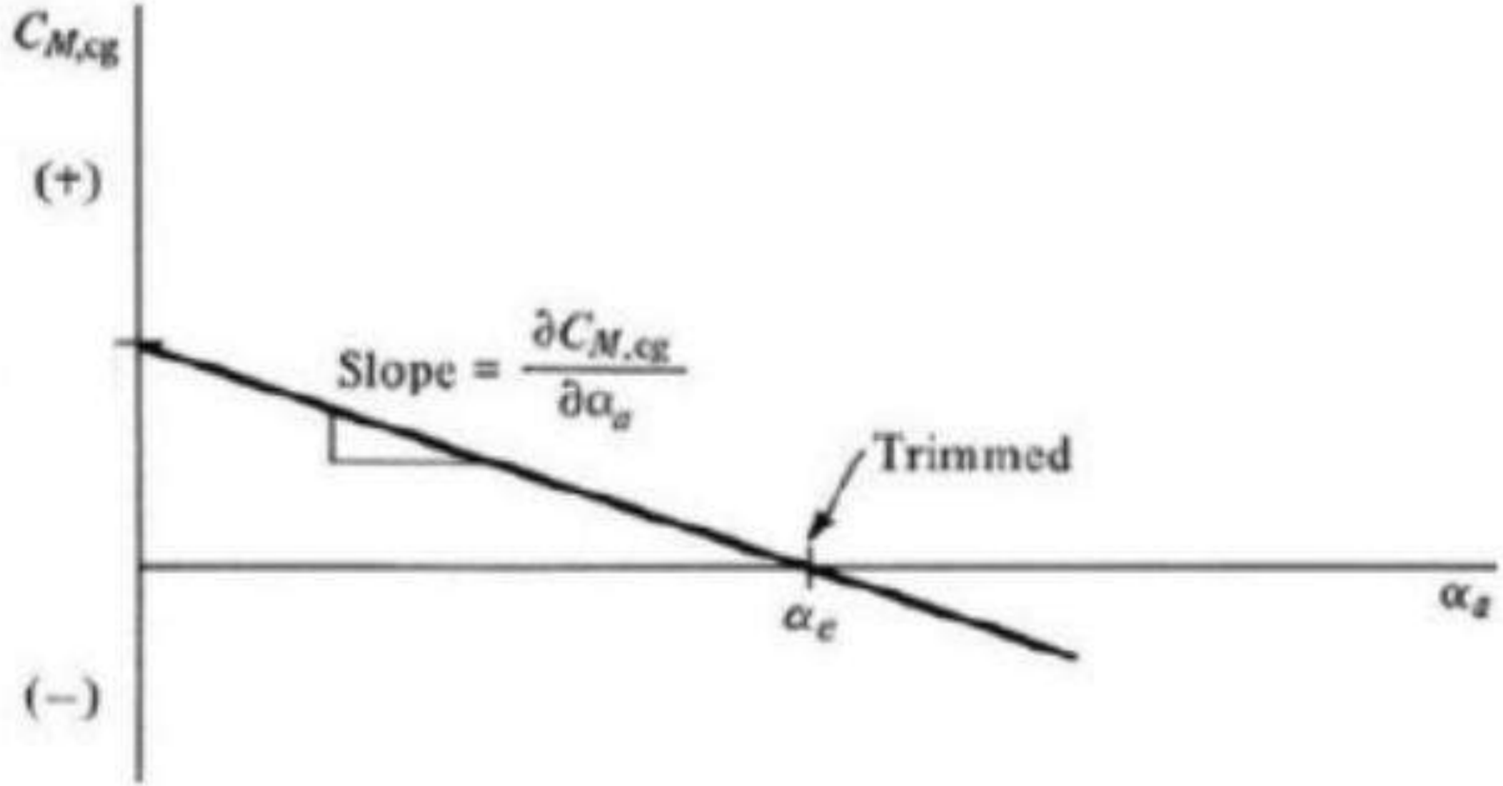
- Consider the complete airplane, as sketched in Fig.
- By examination of Fig., pitching moment about the center of gravity of the airplane *Mcg* is created by
- (1) *L*, *D*, and *Mac* of the wing;
- (2) lift of the tail;
- (3) thrust; and
- (4) aerodynamic forces and moments on other parts of the airplane, such as the fuselage .
- A moment does exist about the center of gravity of an airplane, and this moment is fundamental to the stability and control of the airplane.



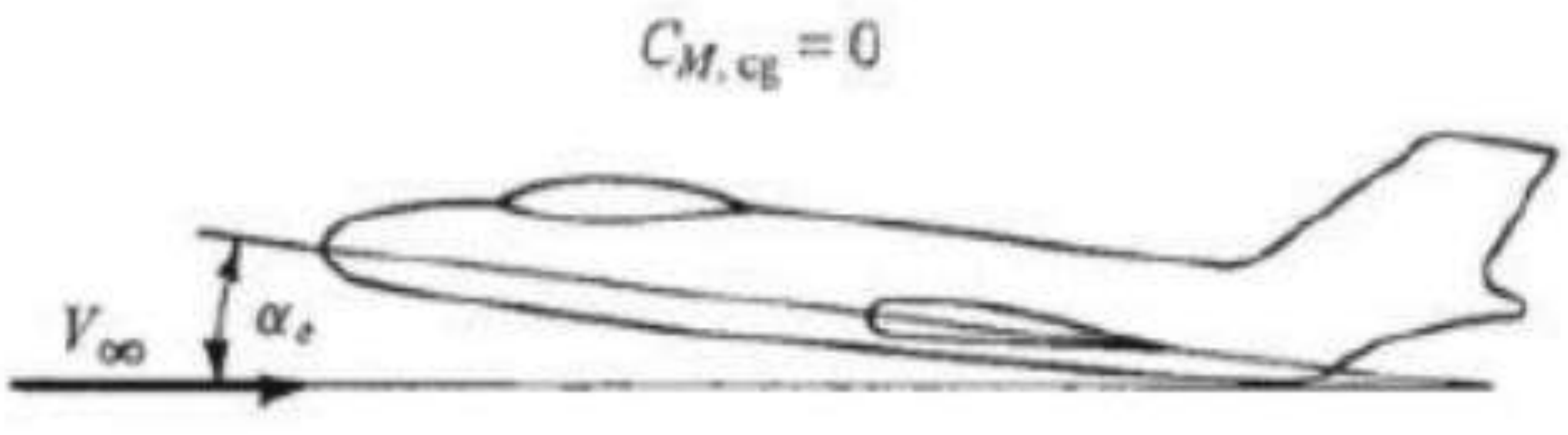
Airplane is in equilibrium when the moment about the center of gravity is zero; that is, when $Mcg = C_{Mcg} = 0$, the airplane is said to be *trimmed.*

LONGITUDINAL STATIC STABILITY

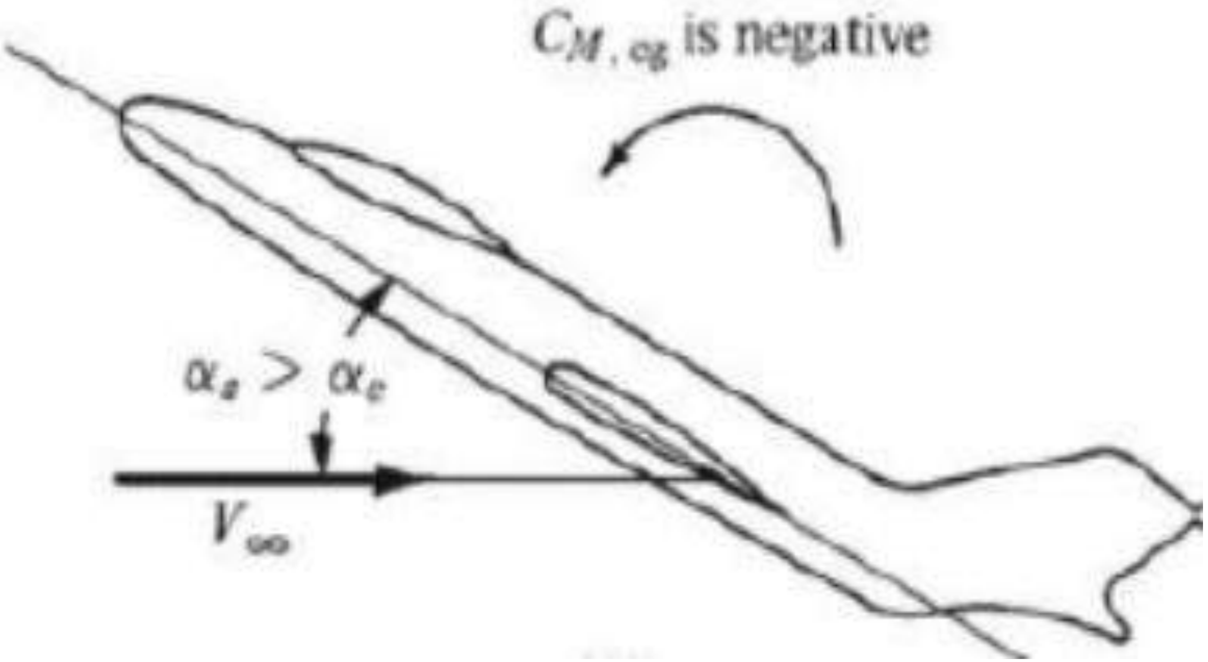
- Consider a rigid airplane with fixed controls, such as the elevator in some fixed position. Assume that the airplane has been tested in a wind tunnel or free flight and that its variation of *Mcg* with angle of attack has been measured. This variation is illustrated in Fig. , where $C_{M,cg}$ is sketched versus α



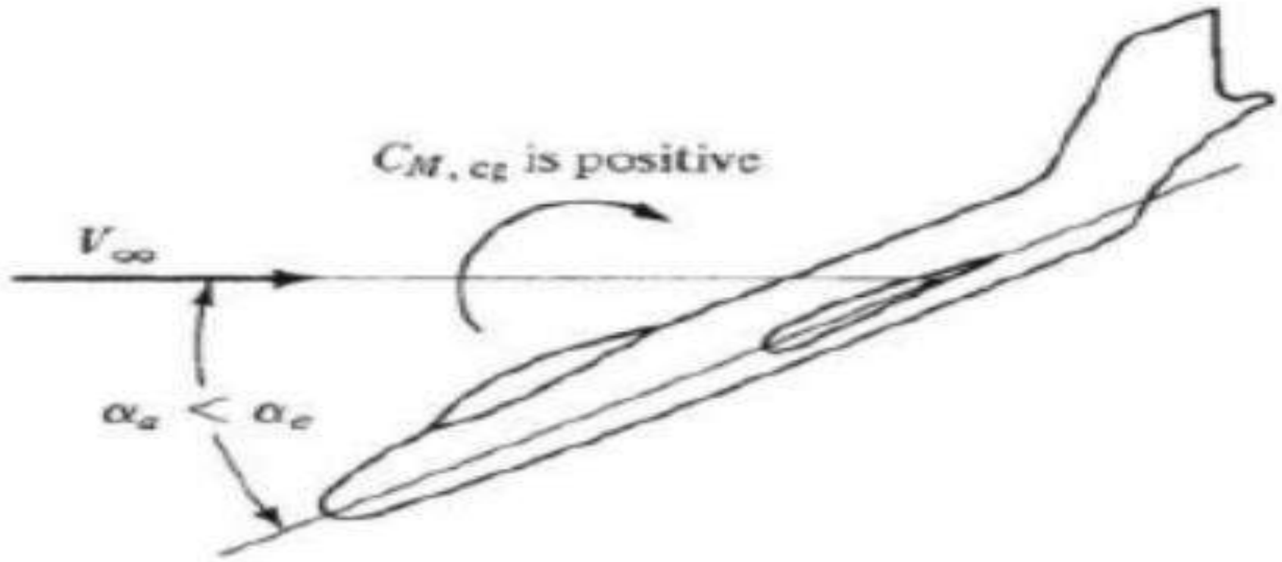
- For many conventional airplanes, this curve is nearly linear, as shown in Fig.
- The value of α where C_{Mcg}= 0 is denoted by α_e , which is the equilibrium, or trim, angle of attack.



- Consider the airplane in steady, equilibrium flight at its trim angle of attack α_e as shown in Fig
- Suddenly the airplane is disturbed by hitting a wind gust, and the angle of attack is momentarily changed.
- There are two possibilities: an increase or a decrease in α_e .
- If the airplane is pitched upward, as shown in Fig. b, then $\alpha.> \alpha_e$
- From Fig. , if $\alpha.> \alpha_e$,the moment about the center of gravity is negative.
- A negative moment (by convention) is counterclockwise, tending to pitch the nose downward. Hence, in Fig. b the airplane will initially tend to move back toward its equilibrium position after being disturbed.


$C_{M,cg}$ is negative
$\alpha_a > \alpha_e$
V_∞

- If the plane is pitched downward by the gust, as shown in Fig. c, then $\alpha_{.} < \alpha_e$.
- From Fig., the resulting moment about the center of gravity will be positive (clockwise) and will tend to pitch the nose upward. Thus we again have the situation in which the airplane will initially tend to move back toward its equilibrium position after being disturbed.
- This is precisely the definition of static stability
- Therefore, an airplane that has a C_{mcg} Vs α . variation like that shown in Fig. 7.13 is *statically stable.*



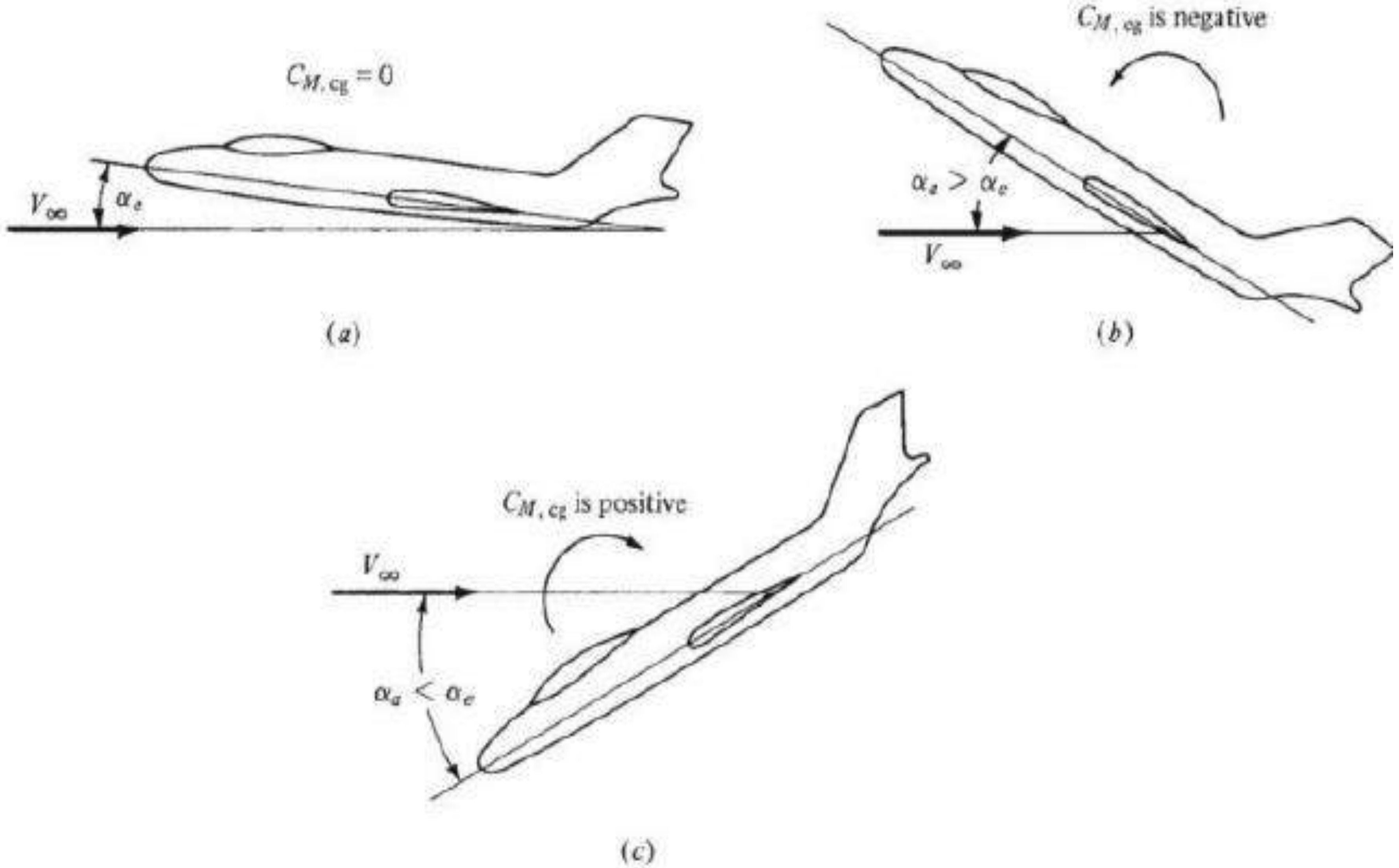


Figure 7.14 Illustration of static stability. (*a*) Equilibrium position (trimmed). (*b*) Pitched upward by disturbance. (*c*) Pitched downward by disturbance. In both (*b*) and (*c*) the airplane has the initial tendency to return to its equilibrium position.

- Consider now a different airplane with a measured CM,cg variation as shown in Fig.
- Imagine that the airplane is flying at its trim angle of attack

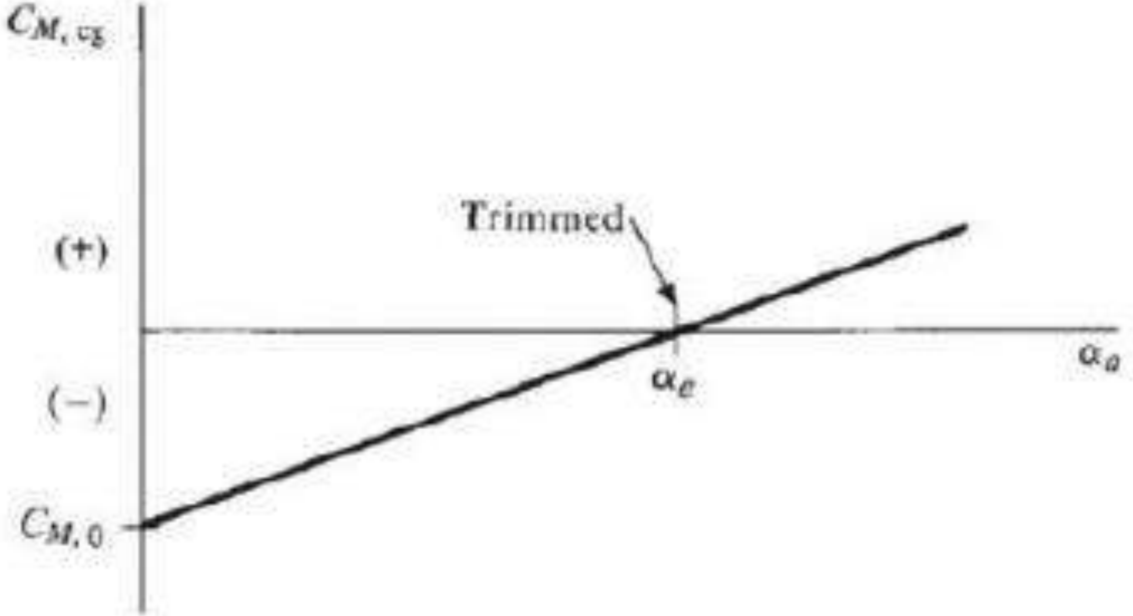
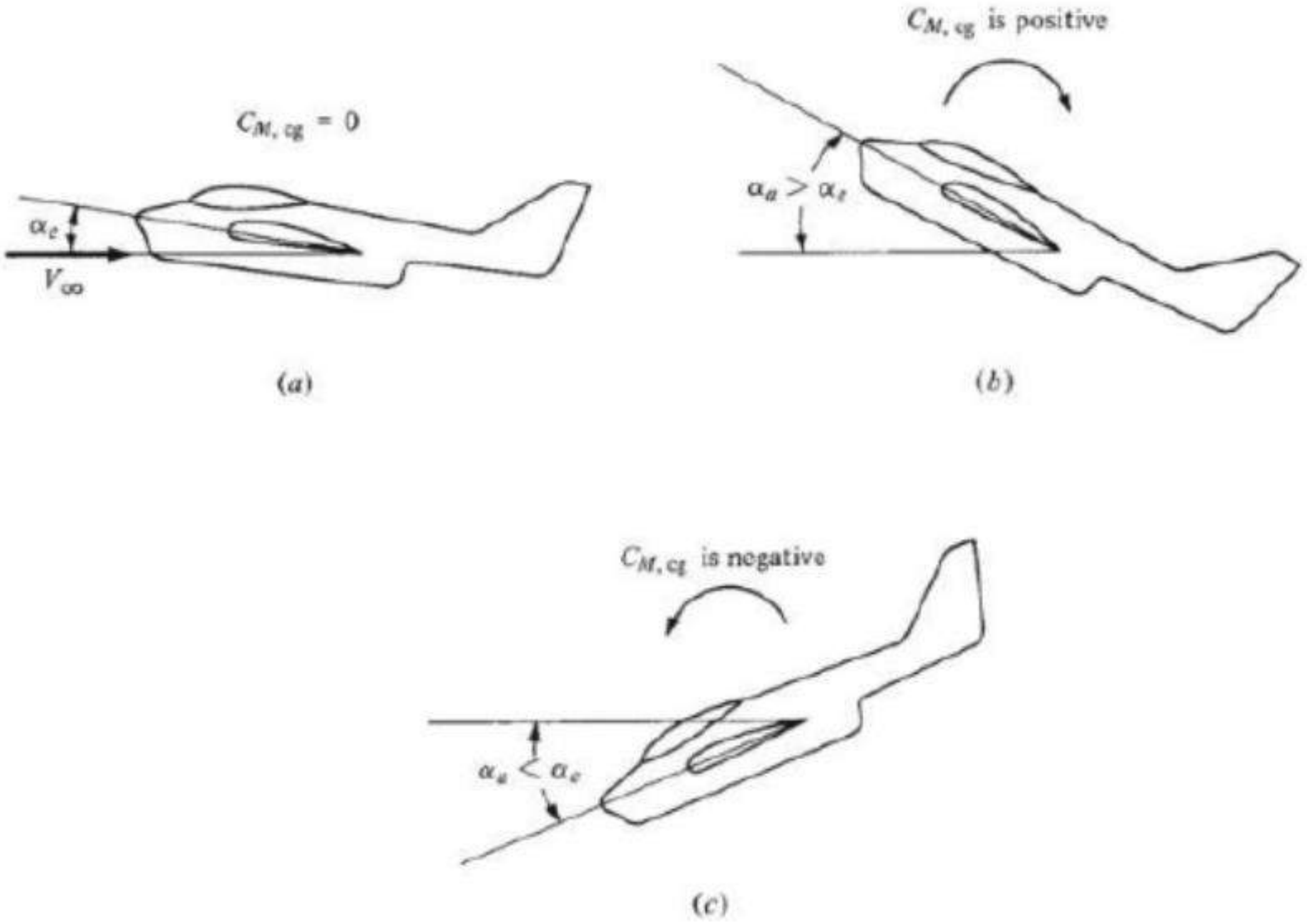


Figure 7.15 Moment coefficient curve with a positive slope.

- If it is disturbed by a *gust*, pitching the nose upward as shown in Fig. b, then $\alpha. > \alpha_e$
- This results in a positive (clockwise) moment, which tends to pitch the nose even further from its equilibrium position. Similarly, if the *gust* pitches the nose downward (Fig. c), a negative (counterclockwise) moment results, which also tends to pitch the nose further from its equilibrium position.
- Therefore, because the airplane always tends to diverge from equilibrium when disturbed, it is *statically unstable.*

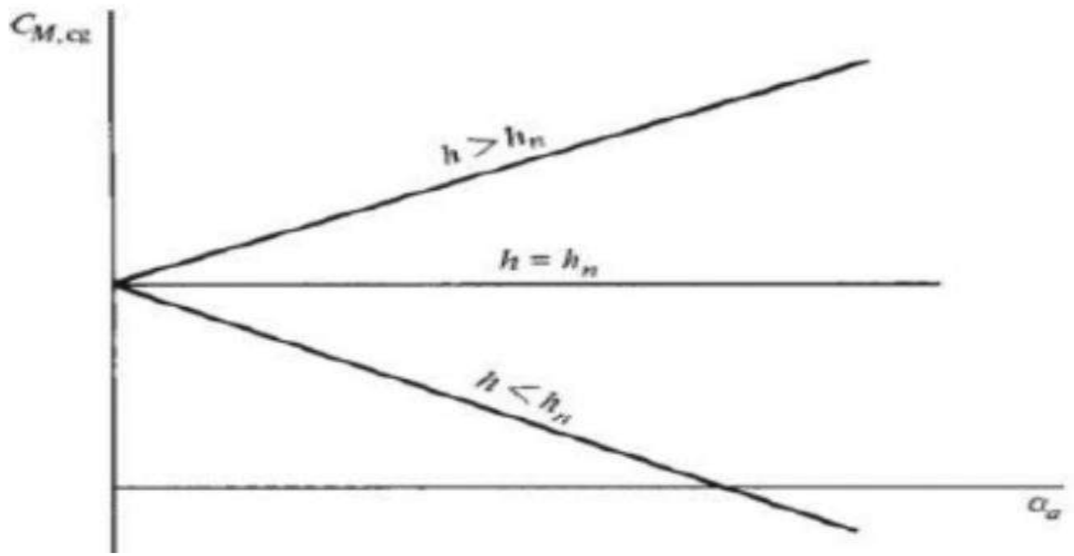

$C_{M,\,cg} = 0$
α_e
V_∞
(a)
$C_{M,\,cg}$ is positive
$\alpha_a > \alpha_e$
(b)
$C_{M,\,cg}$ is negative
$\alpha_a < \alpha_e$
(c)

- The necessary criteria for longitudinal balance and static stability are

1. Slope of the $C_{M,cg}$ versus α curve must be negative.
2. α must fall within the flight range of angle of attack for the airplane.

NEUTRAL POINT

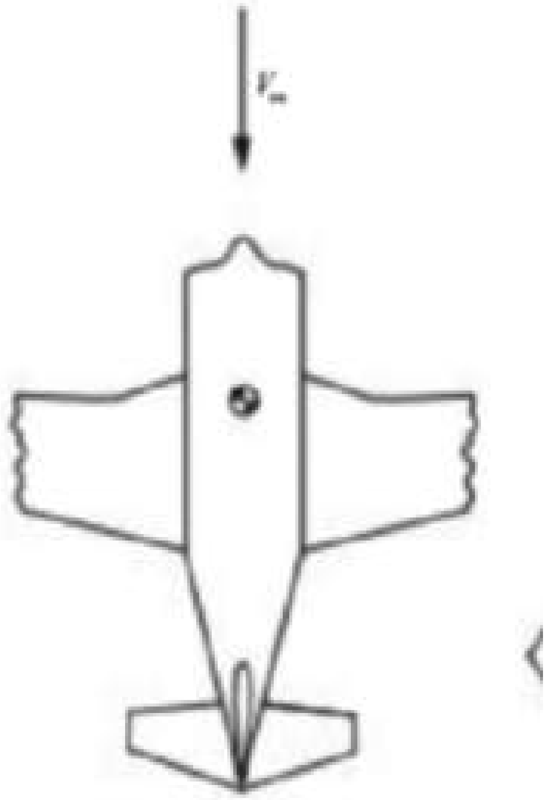
- Consider the situation where the location *h* of the center of gravity is allowed to move with everything else remaining fixed.
- There is one specific location of the center of gravity such that the slope of $C_{M,cg}$ versus α curve is zero.
- The value of *h* when this condition holds is defined as the *neutral point,* denoted by h_n. When $h = h_n$, the slope of the moment coefficient curve is zero.
- For a *given* airplane design, the neutral point is a .*fixed quantity*



- $h < h_n$ means that the center of gravity is located *forward* of the neutral point.
- Thus, an alternative stability criterion is as follows:
- For longitudinal static stability, the position of the center of gravity must always be forward of the neutral point.

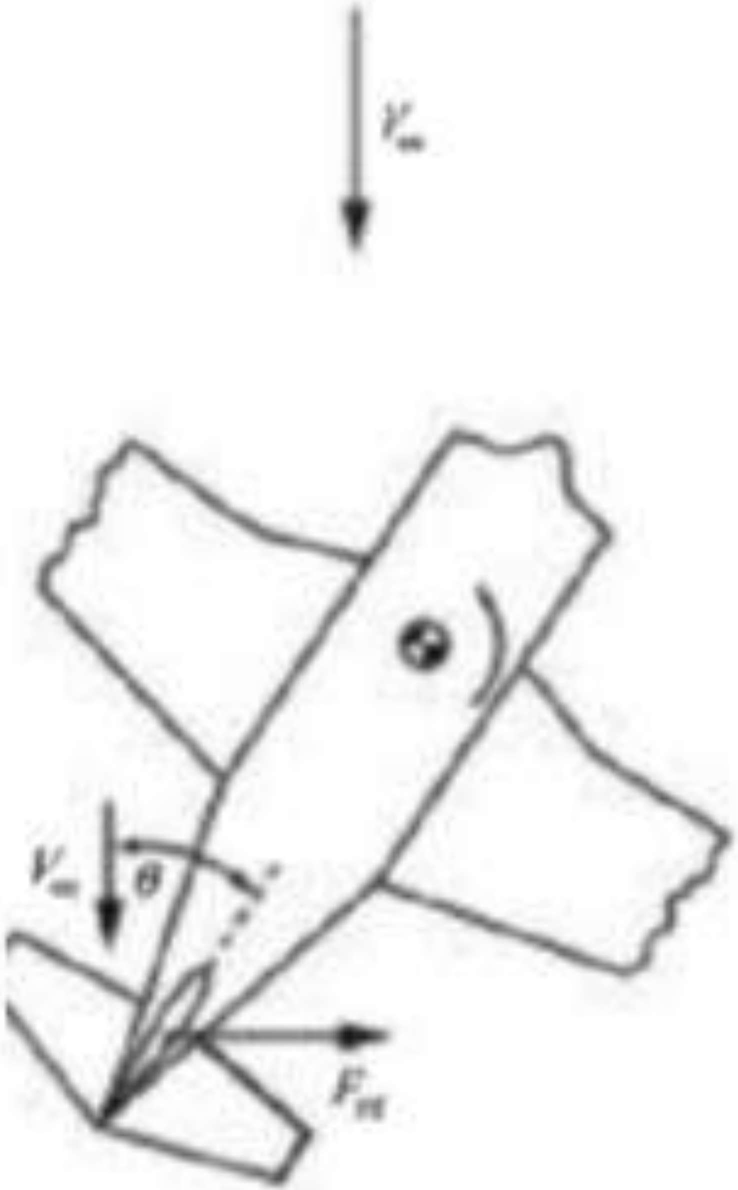
DIRECTIONAL STATIC STABILITY

- Stability in yaw is called *directional stability.*
- Consider an airplane in equilibrium flight with no yaw, as sketched in Fig. . The vertical tail, which is designed with a symmetric airfoil section, is at a zero angle of attack to the free stream, and it experiences no net aerodynamic force perpendicular to *vα*

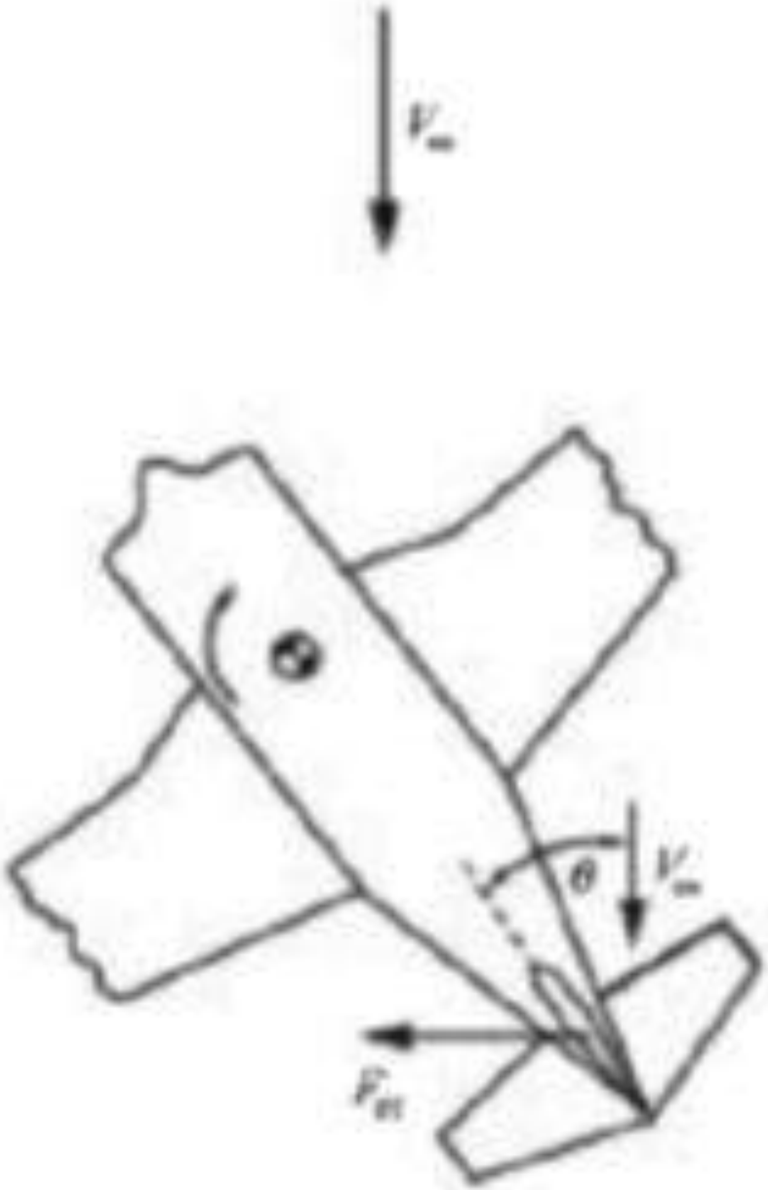


(a) No yaw

- Assume that the airplane is suddenly yawed to the right by a disturbance, as shown in Fig. b.
- The vertical tail is now at an angle of attack Θ and experiences an aerodynamic force F_{vt}, perpendicular to $v\alpha$.
- This force creates a restoring yawing moment about the center of gravity that tends to rotate the airplane back toward its equilibrium position.
- The same situation prevails when the airplane is yawed to the left by a disturbance, as sketched in Fig. c.



(*b*) Yaw to the right



(*c*) Yaw to the left

- The magnitude of the restoring moment in yaw is equal to $F_{vt}l_{vt}$ where l_{vt} is the moment arm from the aerodynamic center of the vertical tail to the airplane's center of gravity, as shown in Fig.

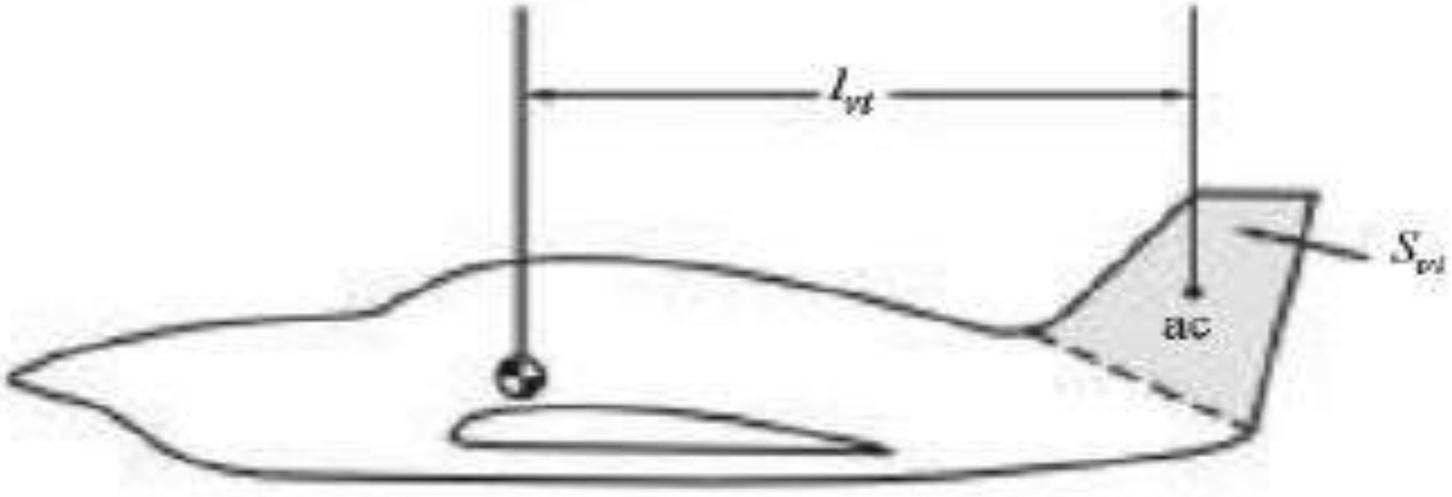


Figure 7.39 Moment arm of the vertical tail.

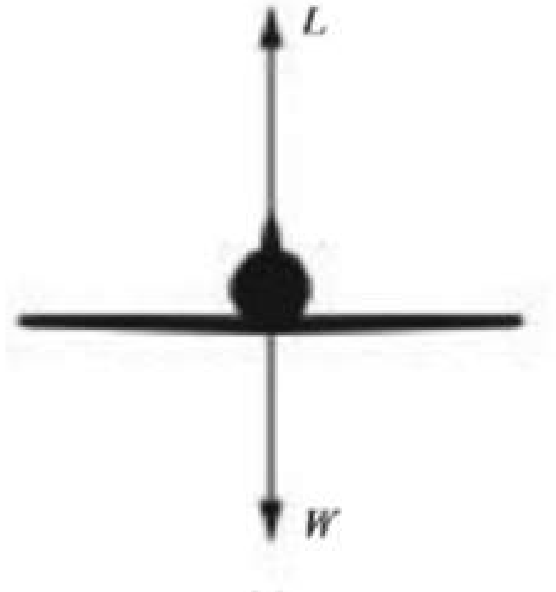
- Because the aerodynamic force on the vertical tail *Fvt* is proportional to the area of the vertical tail S_{vt}, shown as the shaded area in Fig. , the design parameter governing directional stability can be shown to be the vertical tail volume ratio, defined as

$$\text{Vertical tail volume ratio} \equiv V_{vt} \equiv \frac{l_{vt} S_{vt}}{bS}$$

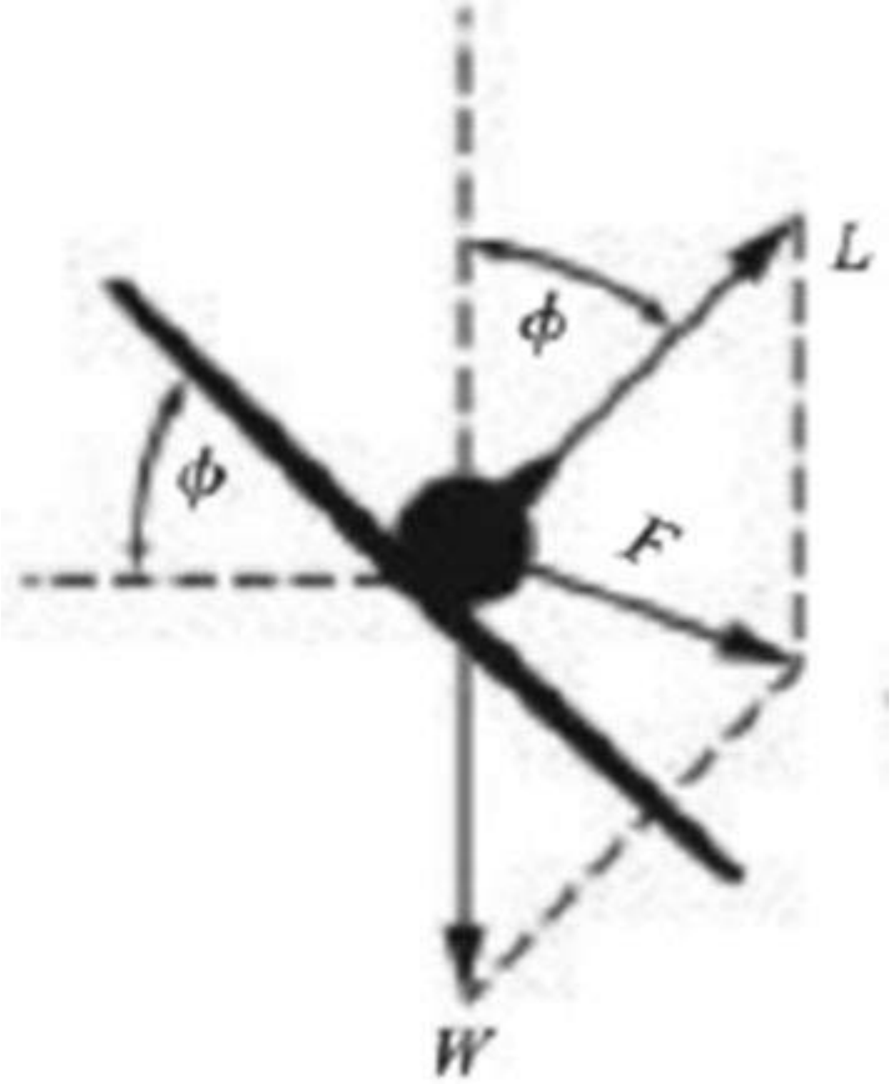
- where b is the wingspan and S is the wing planform area.

LATERAL STATIC STABILITY

- Consider an airplane in steady, level flight. Let us take a view of this airplane from behind, looking in the direction of flight, as sketched in Fig. *a*. The lift equals the weight. They act equal and opposite to each other; there is no net side force.

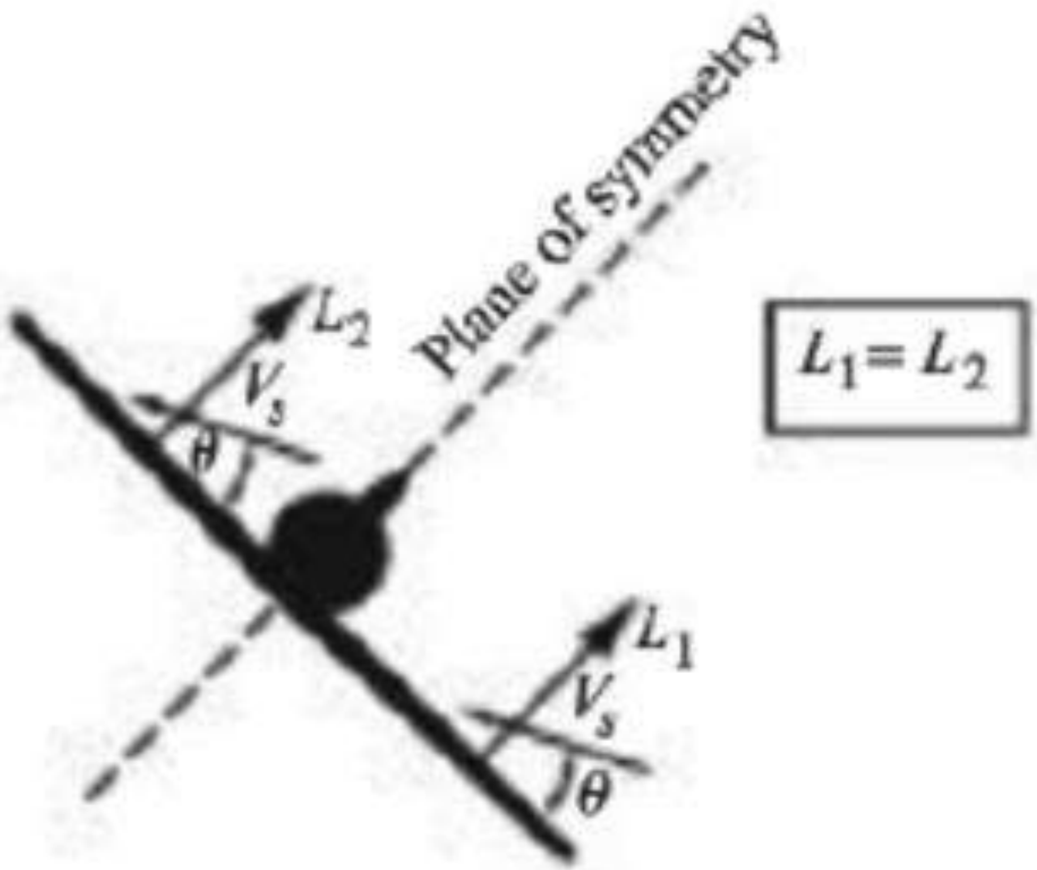


- The airplane is suddenly perturbed by a gust that causes the right wing to dip; that is, a roll to the right. This is sketched in Fig. *b*.
- The lift vector is now rotated from the vertical through angle Φ, called the *bank angle*.
- The vector resolution of L and W results in a side force F, which causes the airplane to accelerate in the direction of F. This sidewise motion of the airplane is called a *slideslip*.



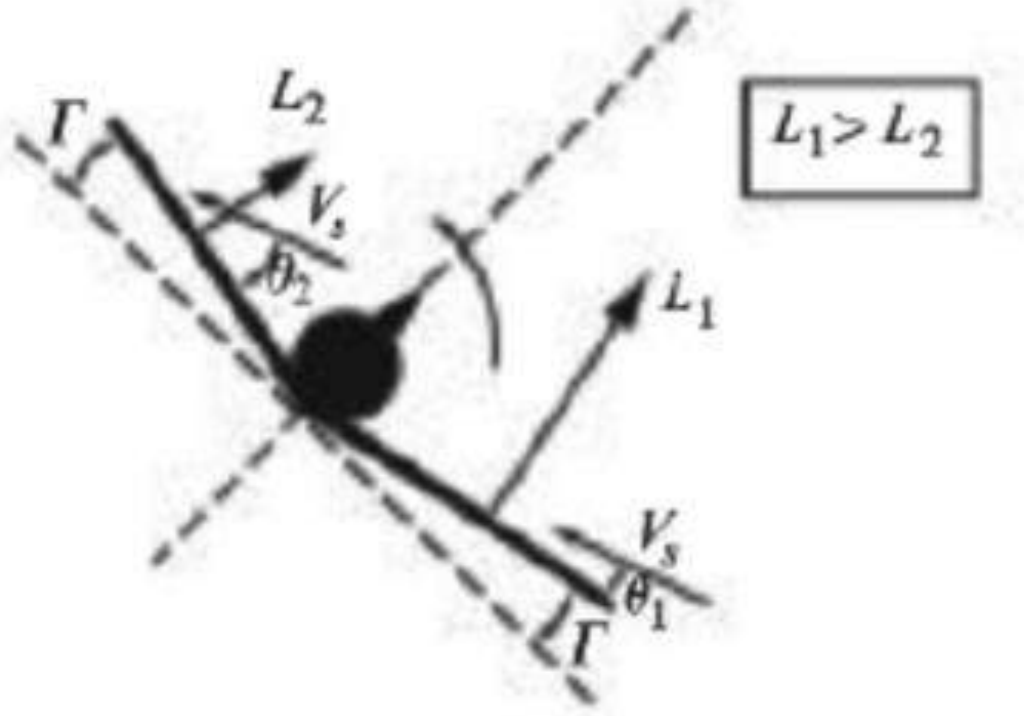
L
ϕ
ϕ
F
W

- Consider the effect of slideslip velocity on the lift generated by the right and left wings.
- Let *L1* and L2 be the lift generated by the right and left wings, respectively. The slideslip velocity *Vs* will affect the lift generated by each wing; but because the two wings are in the same plane, *Vs* makes the same angle Ɵ with respect to both wings; therefore *L1* = L2. as shown in Fig. *a*.
- As a result, there is no restoring moment to return the airplane to its original equilibrium position.



Plane of symmetry
L_2
V_s
θ
L_1
V_s
θ
$L_1 = L_2$

- However, consider the case where both wings are bent upward , as shown in Fig. b; that is, the wings are designed with a V shape. This is called *dihedral,* and the angle is called the dihedral angle.



(b)

- Here the slideslip velocity makes an angle $\Theta_1$with respect to the right wing and a larger angle Θ_2 with respect to the left wing.
- *As* a result, the lift on the left wing *L2* is smaller than the lift on the right wing, and this creates a restoring rolling moment that tends to return the airplane to its equilibrium position, as shown in Fig. *b*.
- Hence, *dihedral is the design feature of the airplane that provides lateral stability.*

Engines

- Modern commercial jet liners have two, or three or even four turbofan engines to provide forward thurst.
- Engines perform with unprecedented reliability, so that in-flight engine failures are very rare, and even if one of the engines fails, on-board computers reconfigure the aircraft to land the aircraft in the nearest airfield.
-

Avionics

- Interdisciplinary
- Avionics of a modern aircraft includes state-of-the-art instrumentation, navigation, communication and warning systems.
- Particularly three warning systems— stall warning system, ground proximity warning systems and traffic collision avoidance system have evolved to a high degree of maturity.

Thank You