X. HIGH SPEED FLIGHT

1) Introduction

When a body is in motion, it will disturb the molecules of air around, those molecules will be displaced in a wave form at the speed of sound, called "pressure wave".

The body in motion, when generating this pressure wave, is moving toward one edge of those pressure waves. At low speed motion, the speed is to slow to be near the pressure waves edge and so the air pressure around is negligibly affected, however at a very high speed, the body will be closer to its pressures waves and the air pressure around is affected significantly due to the compressibility of its pressure waves, called "compressibility effect"

2) Local Speed of Sound

The compressibility effect is assumed to exist when the body in motion reaches 40% of the speed of sound.

Indeed the speed of sound depend on the environment where the wave is traveling. In the air, the speed of sound is affected by the temperature.

Since the temperature decreases with the altitude in a standard atmosphere, therefore the speed of sound changes locally. We speak about the Local Speed of Sound, which will be decreasing with altitude, up to the tropopause where it will remain constant.

At sea level, Local Speed of Sound "a" = 340m/s or 661 kt

$$\boldsymbol{a\ (kt) = 38,95\sqrt{T°C + 273,15}}$$

3) Mach Number

Ratio between the TAS and the Local Speed of Sound a

$$Mach = \frac{TAS}{a}$$

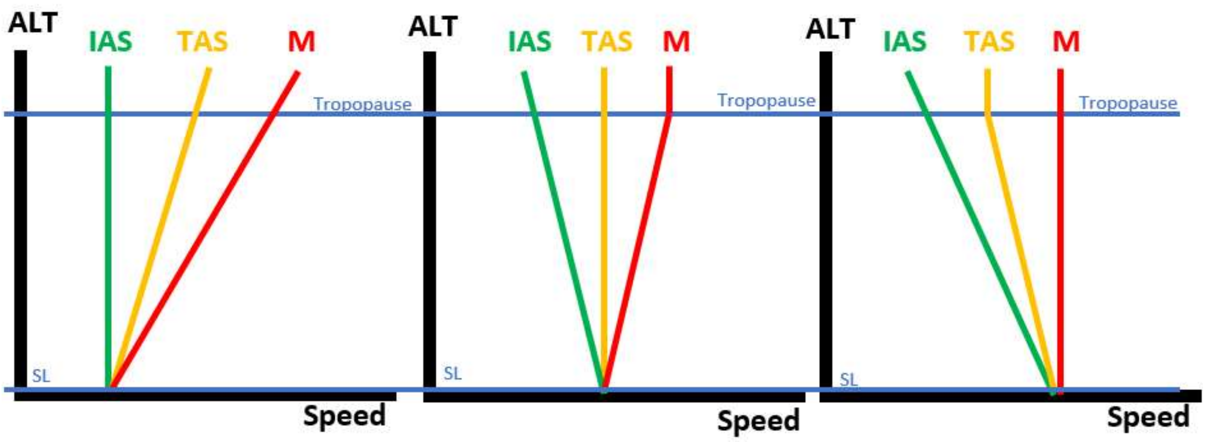
$$Mach = \frac{TAS}{38{,}95\sqrt{T°C + 273{,}15}}$$

Subsonic : M<1.0

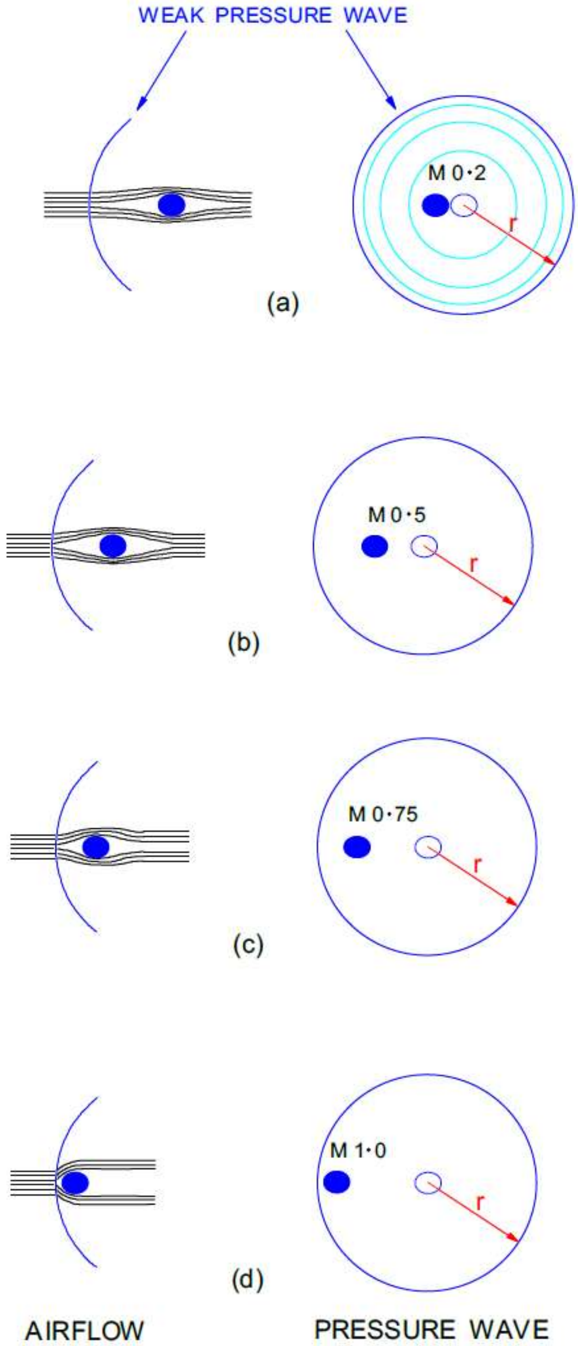
Sonic : M=1.0

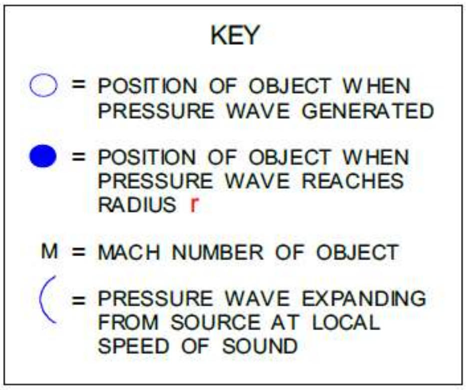
Supersonic : M>1.0

Transonic: $M_{CRIT} \leq M \leq 1.2$



4) Propagation of the Pressure Waves





This series of sketches illustrate the basic idea of pressure wave formation ahead of an object moving at various Mach numbers and of the airflow as it approached the object. Pressure waves are propagated continuously, but for clarity just one is considered.

If we assume a constant local speed of sound; as the object's Mach number increases, the object gets closer to the 'leading edge' of the pressure wave and the air receives less and less warning of the approach of the object.

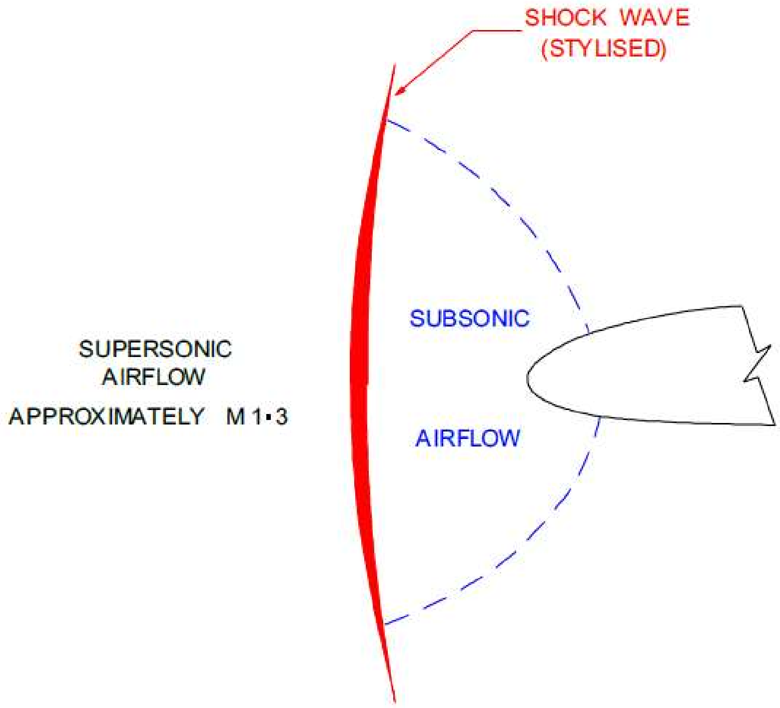
The greater the Mach number of the object, the more acute the upwash angle and the fewer the number of air particles that can move out of the path of the object. Air will begin to build up in front of the object and the density of the air will increase.

When the object's speed has reached the local speed of sound (d), the pressure wave can no longer warn the air particles ahead of the object because the object is travelling forward at the same speed as the wave.

Therefore, the free stream air particles are not aware of anything until the particles that are piled up right in front of the object collide with them. As a result of these collisions, the air pressure and density increase accordingly.

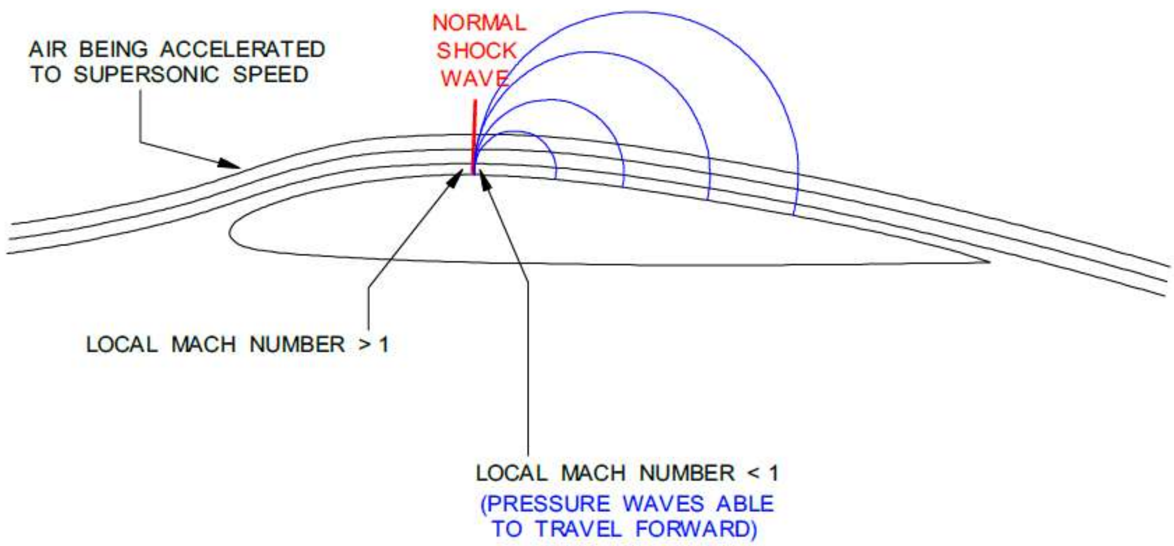
As the object's speed increases to just above M 1.0, the pressure and density of the air just ahead of it are also increased. The region of compressed air extends some distance ahead of the object, the actual distance depends on the speed and size of the object and the temperature of the air.

At one point the free air stream particles are completely undisturbed, having received no advance warning of the approach of a fast moving object, and then are suddenly made to undergo drastic changes in velocity, pressure, temperature and density. Because of the sudden nature of these changes, the boundary line between the undisturbed air and the region of compressed air is called a ‘shock wave’



5) Normal Shock Wave & Critical Mach Number

(Normal meaning perpendicular to the upstream flow). In addition to the formation of a shock wave described overleaf, a shock wave can be generated in an entirely different manner when there is no object in the supersonic airflow. Remember, the airflow accelerates over the aerofoil, meaning that the airflow can reach M1.0 even if the aircraft is traveling at a subsonic speed. Whenever supersonic airflow is slowed to subsonic speed without a change in direction, a ‘normal’ shock wave will form as a boundary between the supersonic and subsonic region. This means that some ‘compressibility effects’ will occur before the whole aircraft reaches Mach 1.0.



CRITICAL MACH NUMBER

An aerofoil generates lift by accelerating air over the top surface. At small angles of attack the highest local velocity on an aircraft will usually be located at the point of maximum thickness on the wing. For example, at a free stream speed of M0.84, maximum local velocity on the wing might be as high as M1.05 in cruising level flight. At increased angles of attack the local velocity will be greater and further forward, also if the thickness/chord ratio were greater the local speed will be higher.

As the free stream speed increases the maximum speed on the aerofoil will reach the local speed of sound first. The Free Stream Mach number at which the local velocity first reaches Mach 1.0 (sonic) is called the Critical Mach number (M_{CRIT}).

Increased thickness/chord and increased angle of attack cause greater accelerations over the top surface of the wing, so the critical Mach number will decrease with increasing thickness/chord ratio or angle of attack.

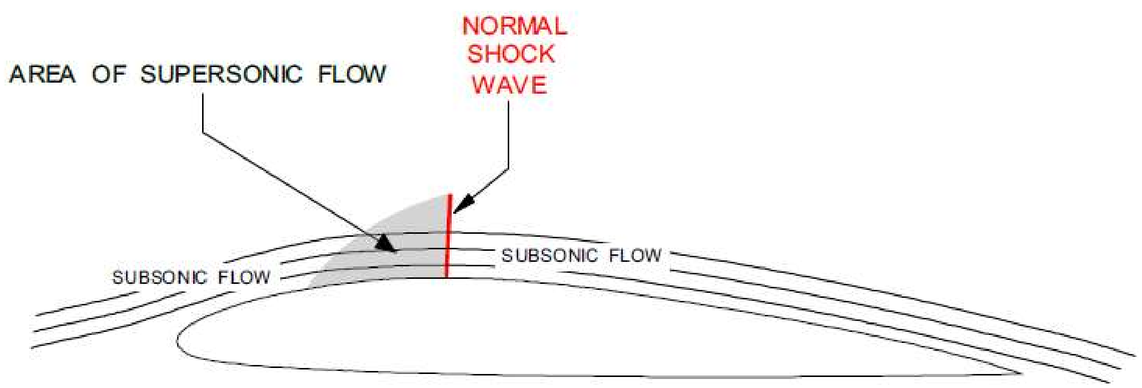
PROPERTIES OF A NORMAL SHOCKWAVE

When a shock wave is perpendicular (normal) to the upstream flow, streamlines pass through the shock wave with no change of direction. A supersonic airstream passing through a normal shock wave will also experience the following changes:

- The airstream is slowed to subsonic
- static pressure increases
- temperature increases
- density increases
- The energy of the airstream [total pressure (dynamic plus static)] is greatly reduced.

6) Accelerating beyond the Critical Mach Number

At speeds just above the critical Mach number there will be a small region of supersonic airflow on the upper surface, terminated by a shock wave

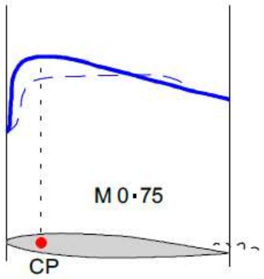


As the aircraft speed is further increased the region of supersonic flow on the upper surface extends and the shockwave marking the end of the supersonic region, moves rearwards. A similar sequence of events will occur on the lower surface although the shockwave will usually form at a higher aircraft speed because the lower surface usually has less curvature so the air is not accelerated so much.

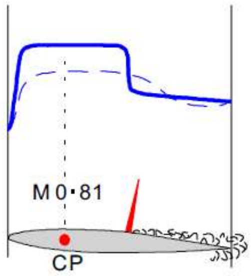
PRESSURE DISTRIBUTION AT TRANSONIC MACH NUMBERS

On the drawings below, the solid blue line represents upper surface pressure and the dashed blue line the lower surface. Decreased pressure is indicated upwards. The difference between the fullline and the dashed line shows the effectiveness of lift production; if the dashed line is above the full line the lift is negative in that area. Lift is represented by the area between the lines, and the Centre of Pressure (CP) by the centre of the area.

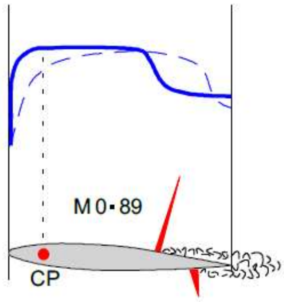
During acceleration to supersonic flight, the pressure distribution is irregular.



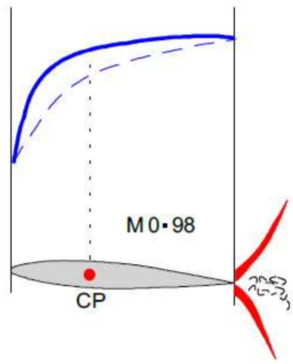
M 0.75 the subsonic picture. Separation has started near the trailing edge and there is practically no net lift over the rear third of the aerofoil section; the CP is well forward.
That CL is quite good and is rising steadily; CD, on the other hand, is beginning to rise.



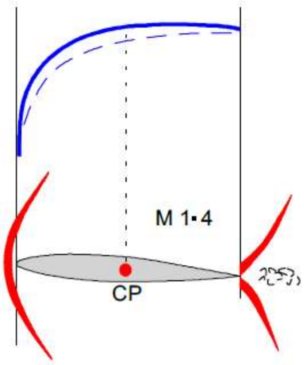
M 0.81 A shock wave has appeared on the top surface; notice the sudden increase of pressure (shown by the falling line) caused by decreasing flow speed at the shock wave. The CP has moved back a little, but the area is still large. The lift is good, but drag is now rising rapidly.



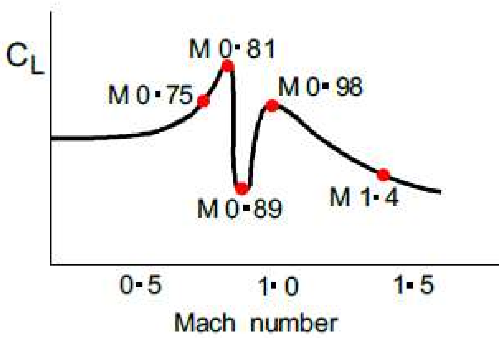
M 0.89 The pressure distribution shows very clearly why there is a sudden drop in lift coefficient before the aerofoil as a whole reaches the speed of sound; on the rear portion of the aerofoil the lift is negative because the suction on the top surface has been spoilt by the shock wave, while there is still quite good suction and high-speed flow on the lower surface. On the front portion there is nearly as much suction on the lower surface as on the upper. The CP has now moved well forward again. The drag is still increasing rapidly.

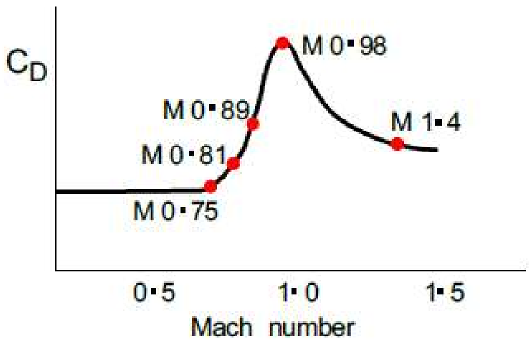


M 0.98 This shows the important results of the shock waves moving to the trailing edge, and no longer spoiling the suction or causing separation. The speed of the flow over the surfaces is nearly all supersonic, the CP has moved aft again, and owing to the good suction over nearly all the top surface, with rather less on the bottom, the lift coefficient has actually increased. The drag coefficient is just about at its maximum.

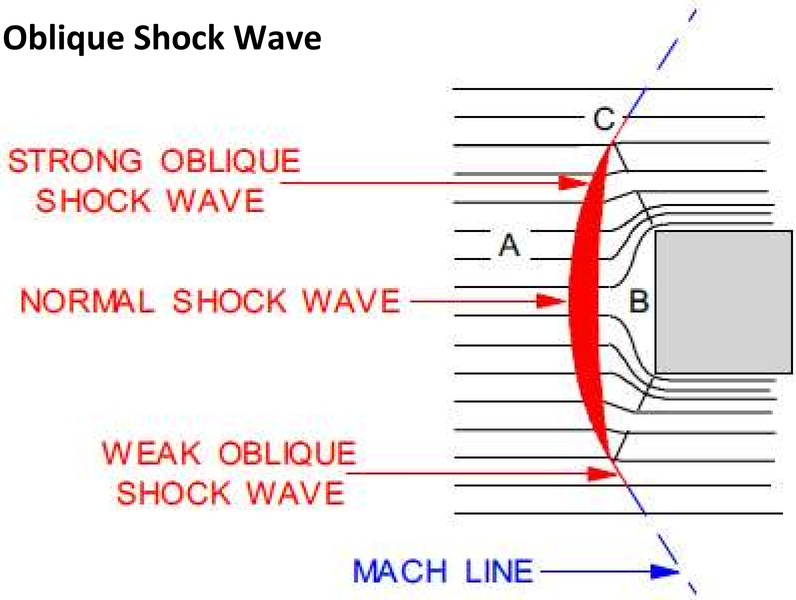


M 1.4 The aerofoil is through the transonic region. The bow wave has appeared. The lift coefficient has fallen again because the pressure on both surfaces are nearly the same; and for the first time since the critical Mach number, the drag coefficient has fallen considerably.





7) Oblique Shock Wave



An oblique shock wave is a slightly different type of shock wave.

Referring to the Figure above, at 'A' the air is travelling at supersonic speed, completely unaware of the approaching object.

The air at 'B' has piled up and is subsonic, trying to slip around the front of the object and merge with the airflow.

Through the shock wave supersonic air from 'A' slows immediately, increasing in pressure and density as it does so. As previously pointed out, a rise in temperature also occurs. The centre part of the shock wave, lying perpendicular or normal to the direction of the airstream, is the strong normal shock wave.

Notice that 'above' and 'below' the normal shock wave, the shock wave is no longer perpendicular to the upstream flow, but is at an oblique angle; the airstream strikes the oblique shock wave and is deflected.

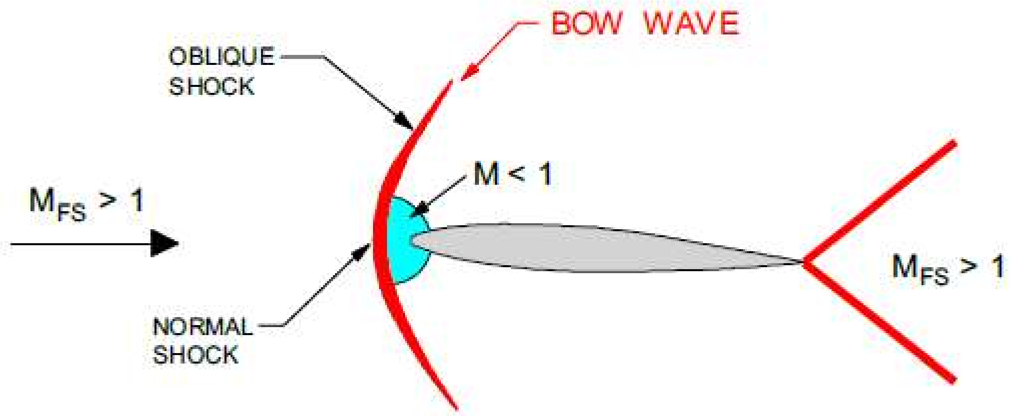
Like the normal shock wave, the oblique shock wave in this region is strong. The airflow will be slowed down; the velocity and Mach number of the airflow behind the wave are reduced, but the flow is still supersonic. The primary difference is that the airstream passing though the oblique shock wave changes direction. (The component of airstream velocity normal to the shockwave will always be subsonic downstream, otherwise no shock wave).

The black dashed lines in Figure outline the area of subsonic flow created behind the strong shock wave.

Particles passing through the wave at 'C' do not slow to subsonic speed. They decrease somewhat in speed and emerge at a slower but still supersonic velocity. At 'C' the shock wave is a weak oblique shock wave. Further out from this point the effects of the shock wave decrease until the air is able to pass the object without being affected. Thus the effects of the shock wave disappears, and the line cannot be properly called a shock wave at all; it is called a 'Mach line'.

8) Bow Wave & Expansion wave

BOW WAVE

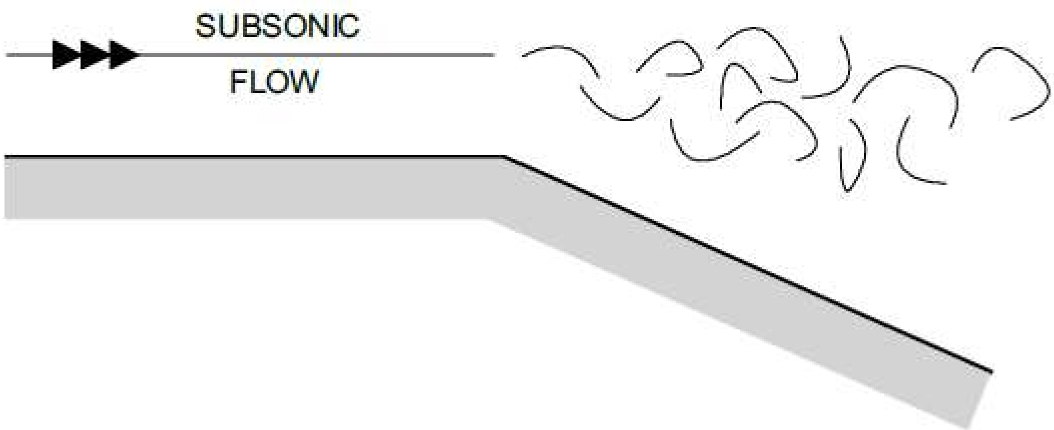


At an aircraft's M>1.0, The shock wave ahead of the leading edge is normal only in the vicinity of the leading edge, and oblique further away from the leading edge ("above" and "below"). This is called a **BOW SHOCKWAVE**

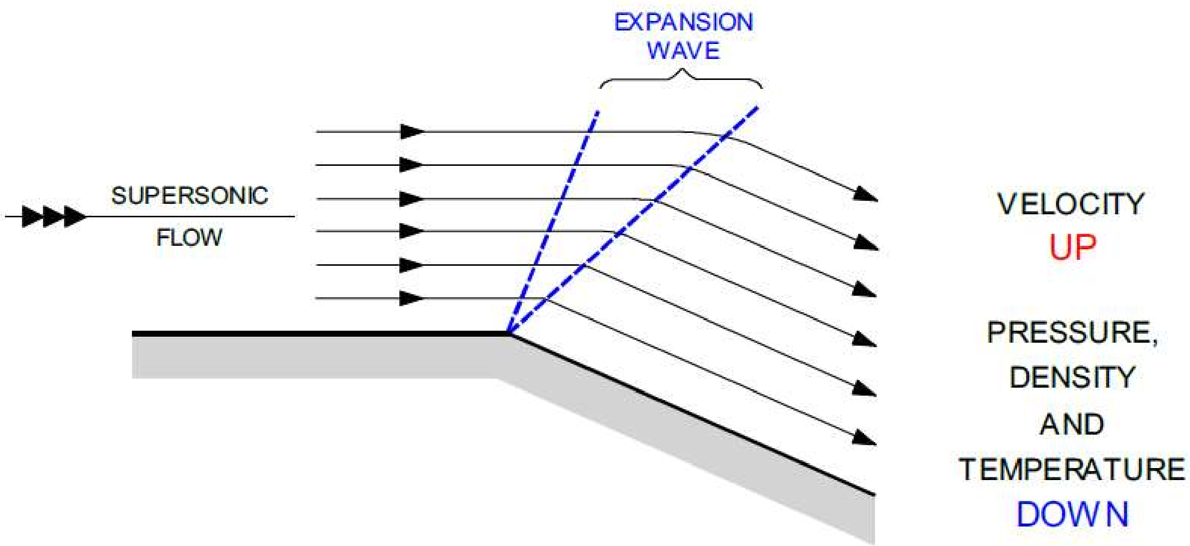
EXPANSION WAVE

In the preceding paragraphs it has been shown that supersonic flow is able to turn a corner by decelerating to subsonic speed when it meets an object. A shock wave forms at the junction of the supersonic and subsonic flow, the generation of which is wasteful of energy (wave drag).

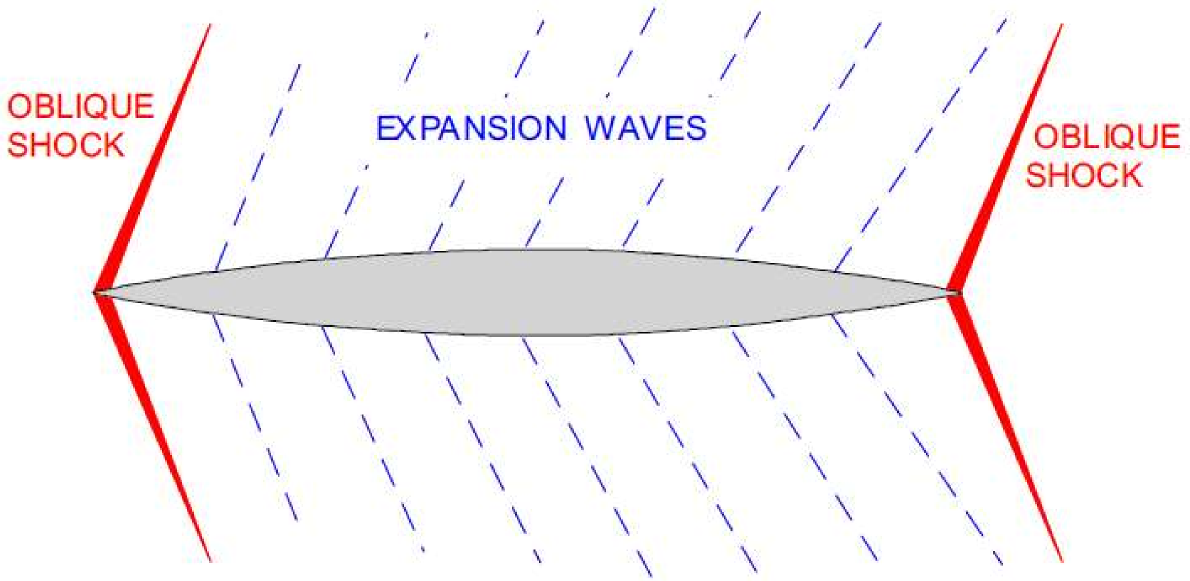
There is another way a supersonic flow is able to turn a corner. Consider first a convex corner with a subsonic flow, as illustrated below:



With subsonic airflow the adverse pressure gradient would be so steep that the airflow would instantly separate at the “corner”.



On the illustration above, it can be seen that a supersonic airflow can follow a convex corner because it expands upon reaching the corner. The velocity INCREASES and the other parameters, pressure, density and temperature DECREASE. Supersonic airflow behaviour through an expansion wave is exactly opposite to that through a shock wave.



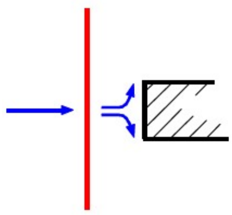
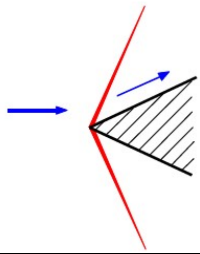
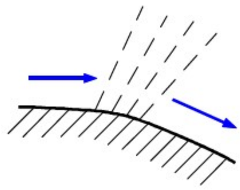
The figure above shows a series of expansion waves in a supersonic airflow. After passing through the bow shock wave, the compressed supersonic flow is free to expand and follow the surface contour. As there are no sudden changes to the airflow, the expansion waves are NOT shock waves.

A supersonic airflow passing through an expansion wave will experience the following changes:

- The airflow is accelerated; the velocity and Mach number behind the expansion wave are greater.
- The flow direction is changed to follow the surface.
- The static pressure of the airflow behind the expansion wave is decreased.
- The density of the airflow behind the expansion wave is decreased.
- Since the flow change is gradual there is no "shock" and no loss of energy in the airflow.

An expansion wave does not dissipate airflow energy.

SUPERSONIC WAVES CHARACTERISTICS

Type of wave	**Normal Shockwave** 	**Oblique Shockwave** 	**Expansion Wave** 
Definition	A plan of discontinuity normal/perpendicular to flow direction	A plan of discontinuity inclined more than 90° from flow direction	
Flow direction change	NO CHANGE	TURN INTO A PRECEEDING FLOW	TURNED AWAY FROM PRECEDING FLOW
Effect of velocity and Mach number behind wave	DECREASED TO SUBSONIC	DECREASED BUT STILL SUPERSONIC	INCREASED TO HIGHER SUPERSONIC
Effect on static pressure ad density	GREAT INCREASE	INCREASE	DECREASE
Effect on energy of airflow	GREAT DECREASE	DECREASE	NO CHANGE (NO CHOCK)
Effect on temperature	GREAT INCREASE	INCREASE	DECREASE

9) Aerodynamic Heating

Air is heated when it is compressed or when it is subjected to friction. An aircraft will have compression at the stagnation point, compression through a shock wave, and friction in the boundary layer.

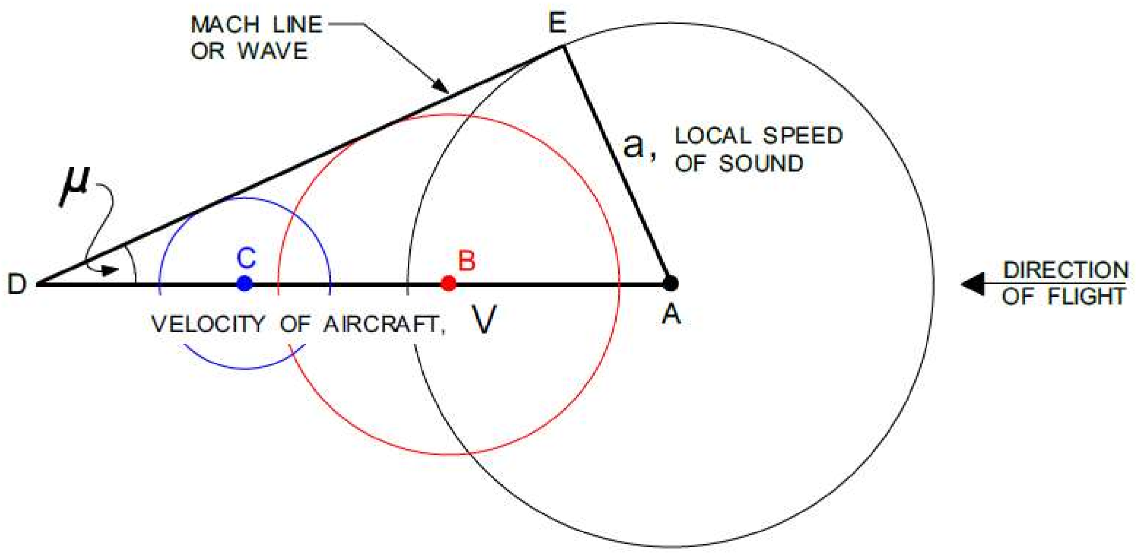
Since the temperature increases **through a shockwave, the Local Speed of Sound on that point increases** and so that's why the airflow Mach number decrease.

So when an aeroplane moves through the air its skin temperature will increase. This occurs at all speeds, but only becomes significant from a skin temperature point of view at higher Mach numbers.

10) Mach Angle & Mach Cone

As the Mach number increases the shock waves become more acute. To illustrate why the angle of the shock waves change it is necessary to consider the meaning and significance of the Mach angle 'µ' (mu).

If the TAS of the aircraft is greater than the local speed of sound, the source of pressure waves is moving faster than the disturbance it creates.



Consider a point moving at velocity 'V' in the direction 'A' to 'D', as in Figure 13.34. A pressure wave propagated when the point is at 'A' will travel spherically outwards at the local speed of sound; but the point is moving faster, and by the time it has reached 'D', the wave from 'A' and other pressure waves sent out when the point was at 'B' and 'C' will have formed circles as shown, and it will be possible to draw a common tangent 'DE' to these pressure waves. The tangent represents the limit to which all the pressure waves have reached when the point has reached 'D'.

'AE' represents the local speed of sound (a) and 'AD' represents the TAS (V)

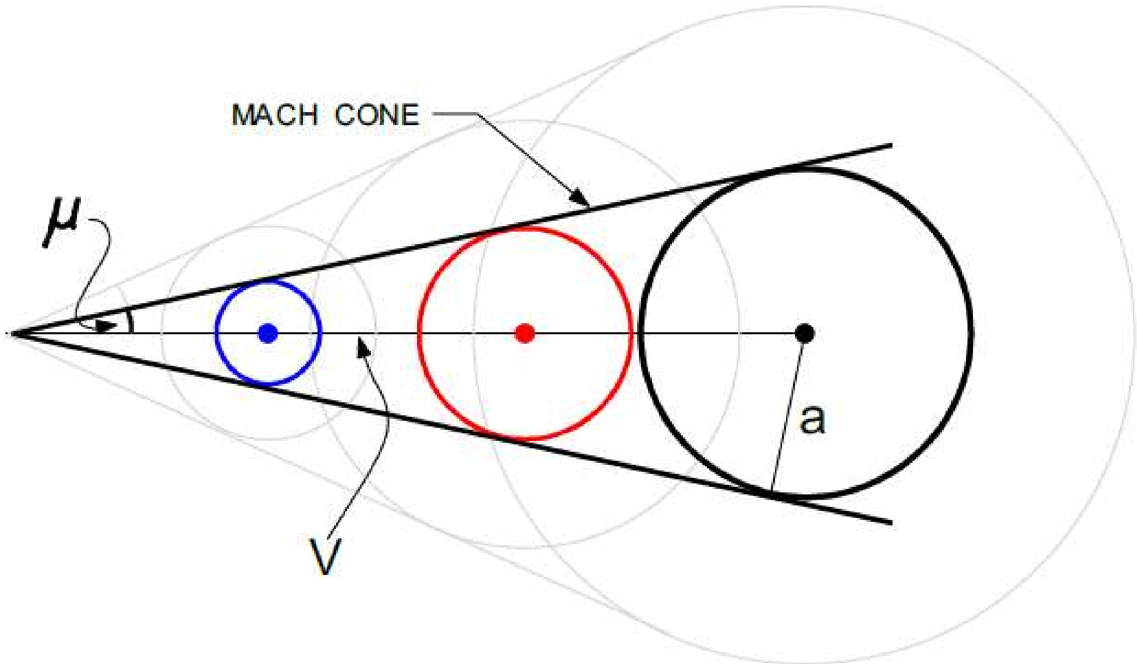
The angle 'ADE', or μ is called the Mach angle and by simple trigonometry:

$$\sin \mu = \frac{a}{TAS} = \frac{1}{M}$$

The greater the Mach number, the more acute the Mach angle μ. At M 1.0, μ is 90°.

MACH CONE

In three dimensions, the disturbances propagating from a moving point source expand outward as spheres, not circles. If the speed of the source (V) is greater than the local speed of sound (a), these spheres are enclosed within a Mach cone, whose semi vertical angle is μ.



It can be seen that the Mach angle (μ) continues to decrease with increasing Mach number. The Mach angle is inversely proportional to the Mach number.

AREA (ZONE) OF INFLUENCE

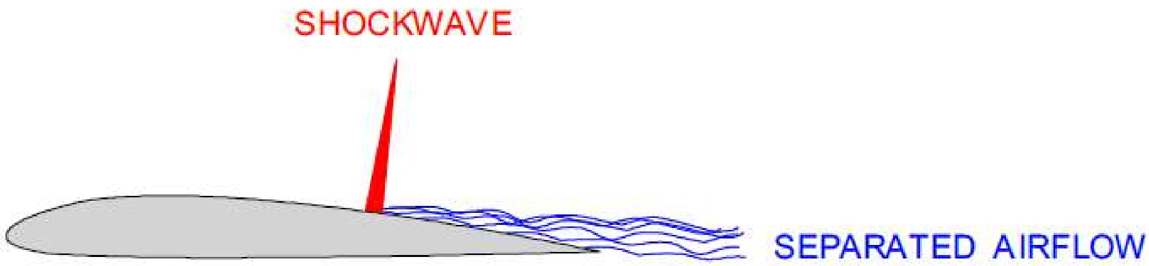
When travelling at supersonic speeds the Mach cone represents the limit of travel of the pressure disturbances created by an aircraft, anything forward of the Mach cone cannot be influenced by the disturbances. The space inside the Mach cone is called the area or zone of influence.

A finite body such as an aircraft will produce a similar pattern of waves but the front will be an oblique shock wave and the wave angle will be greater than the Mach angle because the initial speed of propagation of the shock waves will be greater than the free stream speed of sound.

11) Effect Of Shock wave Formation

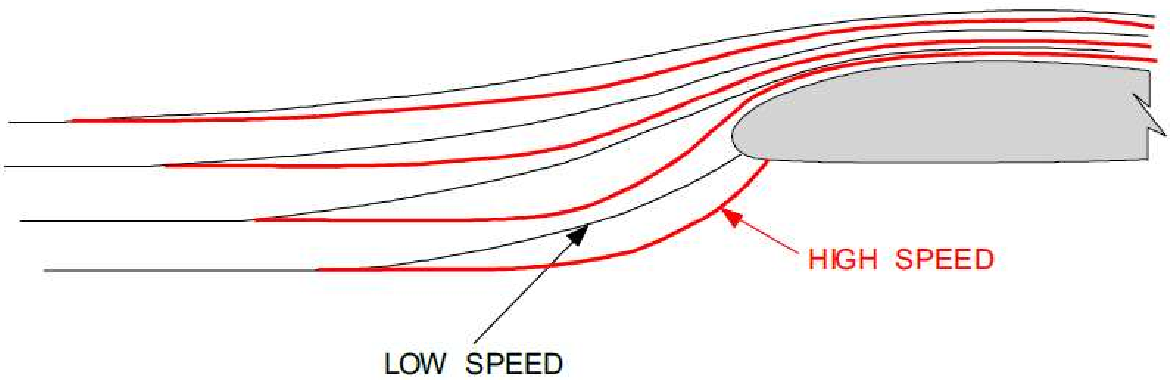
The formation and development of shockwaves on the wing have effects on lift, drag, stability and control. Many of these effects are caused by shock induced separation. As the air flows through the shock wave the sudden rise in pressure causes the boundary layer to thicken and often to separate. This increases the depth of the turbulent wake behind the wing.

At speeds above M_{CRIT} a shockwave will have formed on the upper surface. This may cause boundary layer separation aft of the shock wave, causing loss of lift

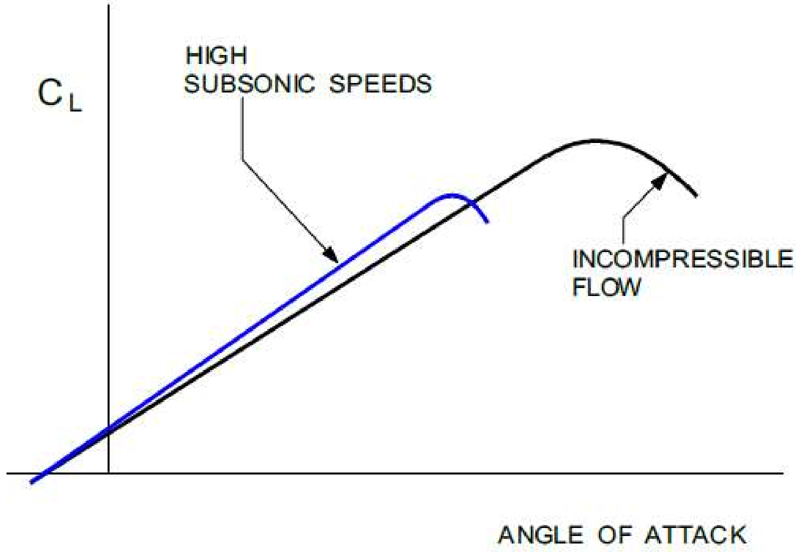


This is known as the **shock stall** because it results from a separated boundary layer just as the low speed stall does. The severity of the loss of lift depends on the shape of the wing sections.
Wings not designed for high speeds may have a severe loss of lift at speeds above M_{CRIT}, but wings designed specifically for high speed flight, with sweep back, thinner sections and less camber will have much less variation of lift through the transonic region.

EFFECT OF SHOCK WAVES ON LIFT CURVE SLOPE AND CL_{MAX}
At high subsonic speed CL increases. This is the result of the changing pattern of streamlines. At low speeds the streamlines begin to diverge well ahead of the aerofoil.
At high subsonic speeds they do not begin to deflect until closer to the leading edge, causing greater acceleration and pressure drop around the leading edge.



At a constant angle of attack, the increase of CL as speed increases from about M 0.4 into the low end of the transonic region gives a steeper lift curve slope, i.e. the change of CL per degree angle of attack will increase. However, because of earlier separation resulting from the formation of the shock wave, C_{LMAX} and the stalling angle will be reduced.



EFFECT OF SHOCK WAVES ON DRAG

As speed increases above M CRIT shock waves begin to form and drag increases more rapidly than it would have done without the shock waves. The additional drag is called wave drag, and is due to energy drag and boundary layer separation.

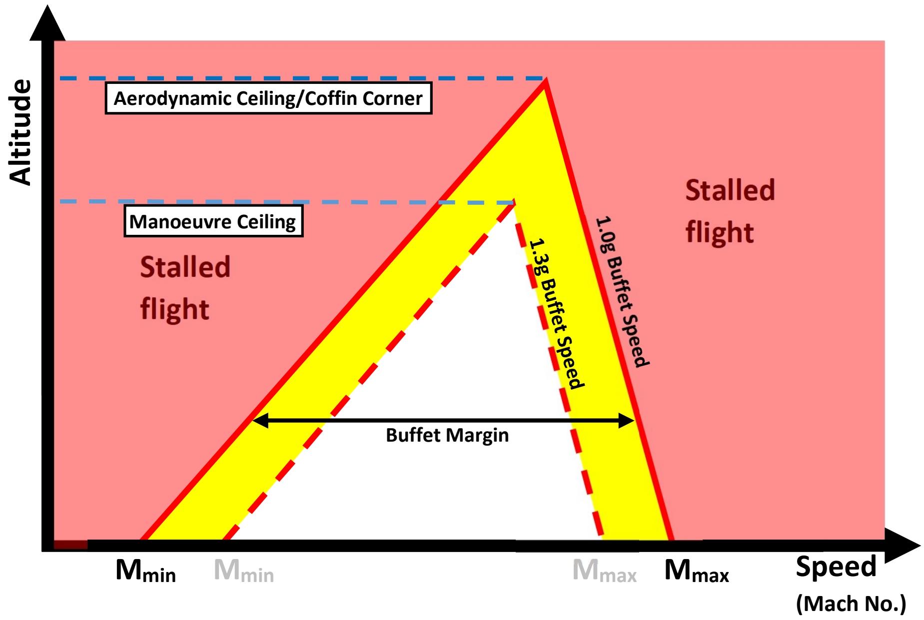
Energy Drag: Energy drag stems from the irreversible nature of the changes which occur as an airflow crosses a shock wave. Energy has to be used to provide the temperature rise across the shock wave and this energy loss is drag on the aircraft. The more oblique the shock waves are, the less energy they absorb, but because they become more extensive laterally and affect more air, the energy drag rises progressively as M_{FS} increases.

Boundary Layer Separation: In certain stages of shock wave movement there is a considerable flow separation. This turbulence represents energy lost to the flow and contributes to the drag. As M_{FS} increases through the transonic range the shock waves move to the trailing edge and the separation decreases; hence the drag coefficient decreases.

12) Buffet & Factors

When the speed of the aeroplane is reduced, to still produce enough lift to balance weight, the angle of attack must increase. However, below a certain speed, the critical angle of attack of the wings is reached, the airflow over the wing separate from the boundary layer producing turbulent airflow. This turbulent airflow buffets on the elevator.
This phenomenon is called **the low-speed buffet**. Flying below this speed will dramatically decrease the lift and a full stall ensues.
For a given weight and configuration, the aeroplane will always stall at the same indicated airspeed but the equivalent Mach number for the low speed buffet and stall increases with altitude
The Mach number for the low speed buffet is abbreviated to $\mathbf{M_{min}}$.

A similar buffet can occur at high speed. At very high speeds, close to the speed of sound, the compressibility of the air ahead of the aeroplane leads to the formation of shockwaves or high pressure waves. These shock waves create a disturbance to the airflow over the wing causing it to separate and create turbulent airflow. Like the low speed buffet, this turbulent airflow will buffet the elevator.
This phenomenon is called **the high-speed buffet** and the Mach number for the high speed buffet decreases with altitude. Flying faster than this speed may cause a high speed shock stall in an aeroplane whose wings are not designed to overcome such effects. The Mach number for the high speed buffet decreases with altitude, as shown.
This speed is commonly abbreviated to $\mathbf{M_{max}}$



Taking into consideration both the Mach number for low speed and high speed buffet, it means that there are two mach numbers, below and above which the aeroplane is unable to fly.
This speed range between the Mach number for the low speed stall and high speed stall is called **the buffet margin.**

The buffet margin is the speed range between the low and speed and high speed buffet. The important point to understand is that the margin between the low speed and high speed buffet decreases with altitude.

Notice that there is an altitude with that the low speed and high speed buffets are coincident at the same velocity. It is impossible to fly higher than this altitude. Flying slower or faster than the speed shown will stall the aeroplane.
This altitude is called **the aerodynamic ceiling**, or **coffin corner**.

In fact, at the aerodynamic ceiling even manoeuvring the aeroplane will initiate a stall because manoeuvring the aeroplane will increase the effective weight (load factor g) and increase the stall speed. To prevent aeroplanes from operating too close to this altitude, an operational limit is set below this point. The aeroplane must manoeuvre up to 40° bank, which will produce:

$$\frac{1}{cos\ 40°} = 1{,}3\ g$$

Aerodynamic ceiling is the altitude where the low speed and high speed buffets are coincident.
Notice that a 1.3 g manoeuvre moves the buffet speed lines to the "dashed red" position.
Notice that now, the Mach number for the low and speed and high speed buffet are coincident at a lower altitude.
This altitude is called the **1.3 g buffet limit altitude** or **manoeuvre ceiling** and is usually about 4,000 to 6,000 ft below the aerodynamic ceiling.

LOAD FACTOR

A higher the load factor requires a higher AoA which will cause the airflow to accelerate further above the aerofoil, causing a shockwave to appear earlier and so an earlier shock stall. In addition, the increase in g causes the stall speed to increase. The buffet margin is then reduced and the aerodynamic ceiling is reduced

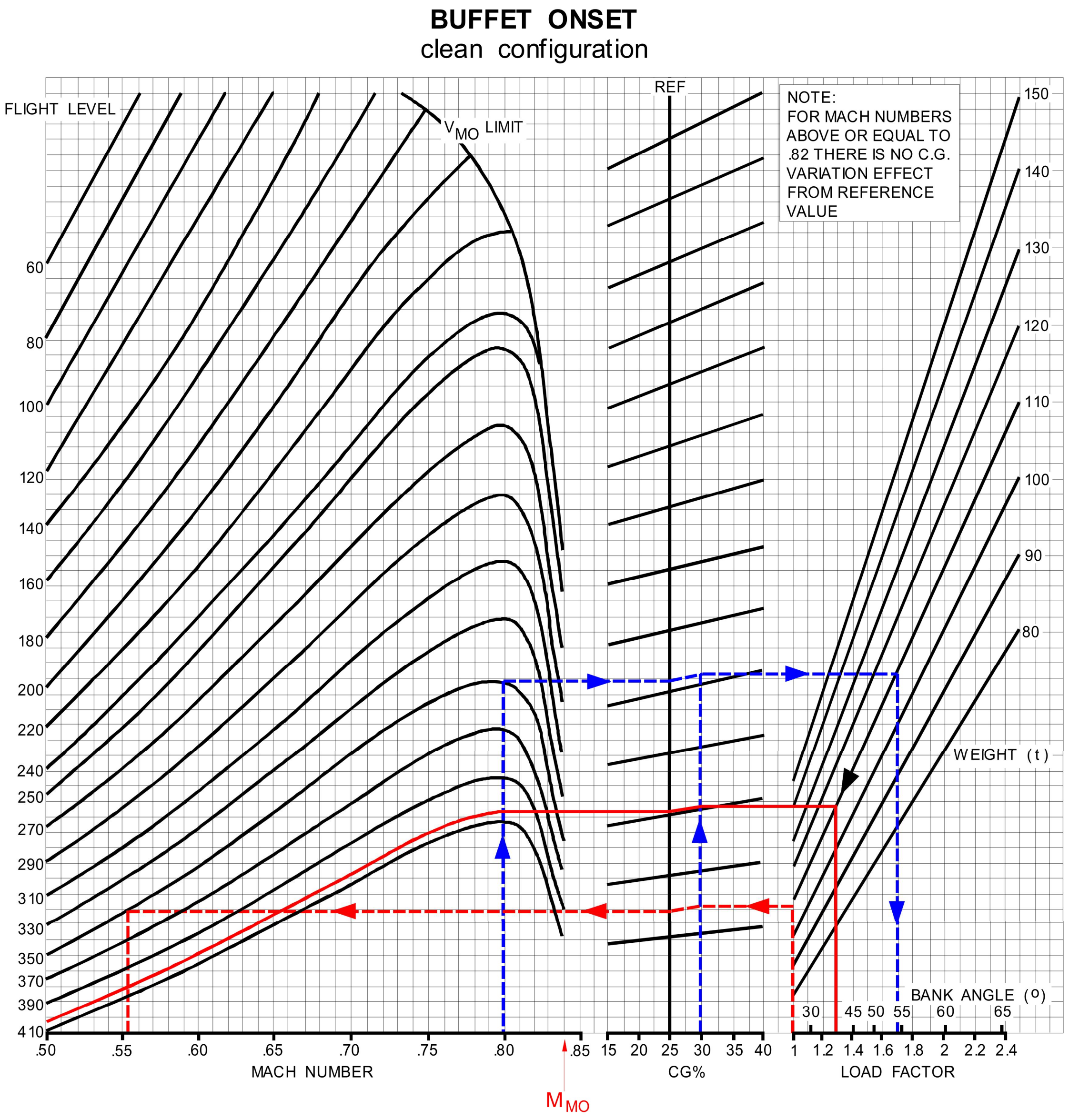
WEIGHT

At a higher weight, the stall speed increase, and the required C_L increase, so the AoA is increased causing the airflow to accelerate further above the aerofoil, causing a shockwave to appear earlier and so an earlier shock stall. The buffet margin is then reduced and the aerodynamic ceiling is reduced

CG POSITION

At forward CG increases the effective weight of the aircraft and the consequences are the same as the weight if the aircraft increases.

13) Use of Buffet Onset Chart



DATA : M = .80
FL = 350
WEIGHT = 110 tons
CG = 30 %

RESULTS : BUFFET ONSET AT :

M = 0.80 WITH 54° BANK ANGLE, OR AT 1.7 g
LOW SPEED (1 g) : M = 0.555
HIGH SPEED : ABOVE M 0.84 (M_{MO})

1.3 g ALTITUDE = FL 405

1.3 g Altitude (1 g + 0.3 g = 1.3 g): At this altitude a 'g' increment of 0,3 can be sustained without buffet occurring. Using the data supplied:-

Follow the vertical solid red line upwards from 1.3 g to the 110 tons line, then horizontally to the 30% CG vertical line, then parallel to the CG reference line, again horizontally to the M0.8 vertical line. The altitude curve must now be 'parallelled' to read-off the Flight Level FL405. The 1.3 g altitude is 40,500 ft.

If the aircraft is operated above FL405 at this mass and CG a gust, or bank angle of less than 40°, could cause the aircraft to buffet. (40° of bank at high altitude is excessive, a normal operational maximum at high altitude would be 10° to 15°).

Buffet restricted speed limits: Using the data supplied:-

Follow the vertical dashed red line upwards from 1 g to the 110 tons line, then horizontally to the 30% CG vertical line, then parallel to the CG reference line. Observe the FL 350 curve. The curve does not reach the horizontal dashed red line at the high speed end because M0.84 (M_{MO})is the maximum operating speed limit. At the low speed end of the dashed red line, the FL 350 curve is intersected at M 0.555. Thus under the stated conditions, the low speed buffet restriction is M 0.555 and there is no high speed buffet restriction because M_{MO} is the maximum operating Mach number which may not be exceeded under any circumstances.

Load factor and bank angle at which buffet occurs: Using the data supplied:-

From M 0.8, follow the dashed blue line to obtain 54° bank angle or 1.7 g.

(Additional example) Aerodynamic ceiling at 150 tons can be determined by:-

Following the vertical dashed red line from 1 g to the 150 tons line, then following the solid red line horizontally to M0.8 (via the CG correction). The altitude curve gives an aerodynamic ceiling of FL390.

14) Delaying or Reducing the effect of compressibility

To maximise revenue, airlines require their aircraft to fly as fast and as efficiently as possible. It has been shown that the formation of shock waves on the wing results in many undesirable characteristics and a massive increase in drag. Up to speeds in the region of M_{CRIT} the effects of compressibility are not too serious. It is therefore necessary to increase M_{CRIT} as much as possible. Many methods have been adopted to delay or reduce the effects of compressibility to a higher Mach number

THIN WING SECTIONS

On a low t/c ratio wing, the flow acceleration is reduced, thus raising the value of M_{CRIT}. For example if M_{CRIT} for a 15% t/c wing is M 0.75, then MCRIT for a 5% t/c wing will be approximately M 0.85.

The use of a low t/c ratio wing section has some disadvantages:

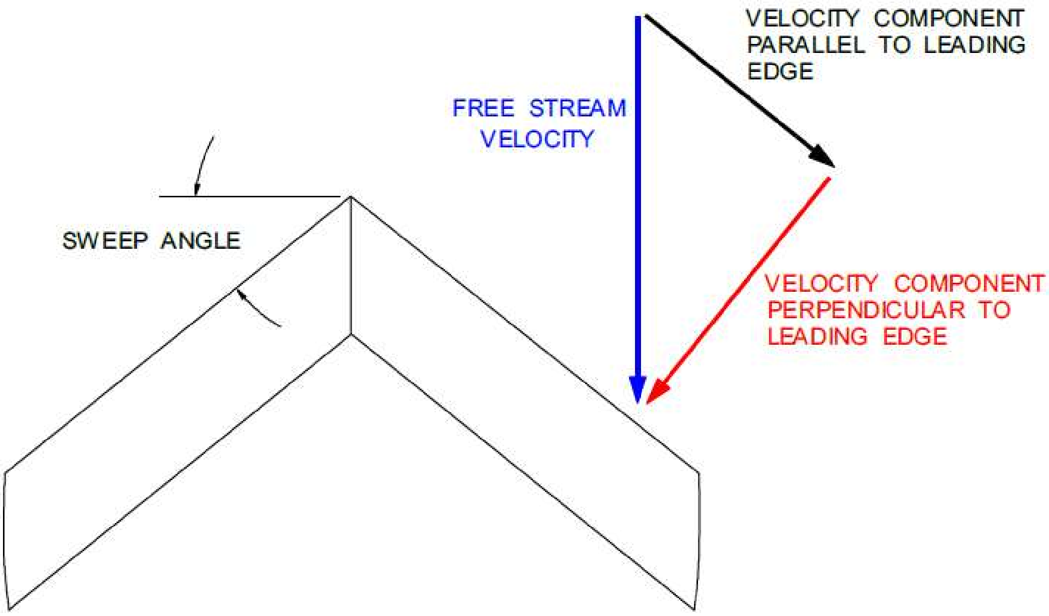
- The lift produced by a thin wing will be less, giving higher take-off and landing speeds and increased distances.
- A thin wing requires disproportionally wider main spars for the same strength and stiffness. This increases structural weight.
- Limited stowage space is available in a thin wing for:
  - Fuel
  - high lift devices and their actuating mechanism and
  - the main undercarriage and its actuating mechanism.

SWEEPBACK

One of the most commonly used methods of increasing MCRIT is to sweep the wing back. Forward sweep gives a similarly effect but wing bending and twisting creates such a problem that sweep back is more practical for ordinary applications.

A simplified method of visualising the effect of sweepback is shown below. The swept wing shown has the free stream velocity broken down to a component of velocity perpendicular to the leading edge and a component parallel to the leading edge.

The component of velocity perpendicular to the leading edge is less than the free stream velocity (by the cosine of the sweep angle) and it is this velocity component which determines the magnitude of the pressure distribution. M_{CRIT} will increase since the velocity component affecting the pressure distribution is less than the free stream velocity.



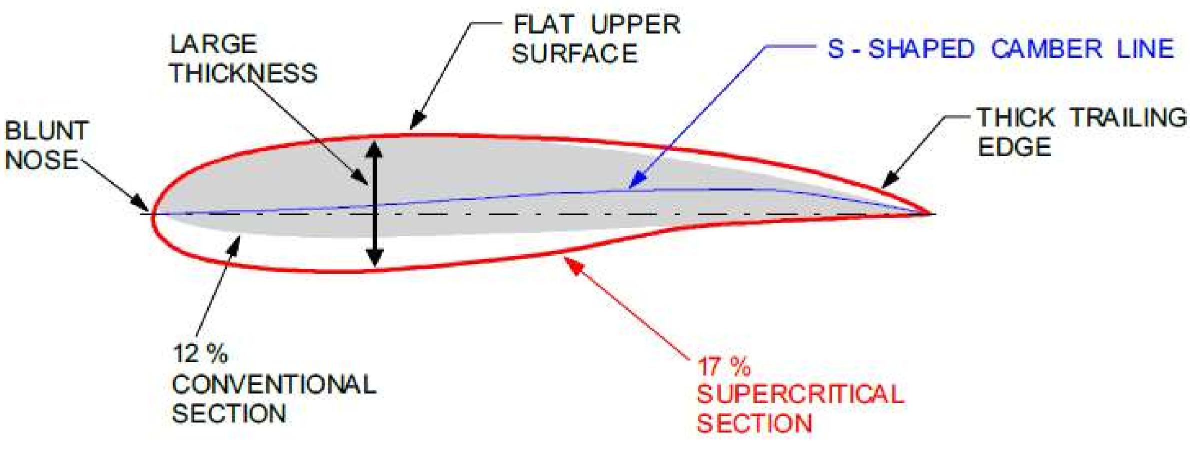
Sweeping the wing back has nearly the same aerodynamic advantages as a thin wing, without suffering reduced strength and fuel capacity.

Unfortunately, there are some disadvantages:

- Reduced C_{LMAX}
  - gives a higher stall speed and increased take-off and landing distances.
  - Maximum lift angle of attack is increased, which complicates the problem of landing gear design (possibility of tail-strike) and reduced visibility from the flight deck during take-off and landing.
- A sweptback wing has an increased tendency to tip stall resulting in pitch-up at the stall and possible deep stall problems.
- Reduced effectiveness of trailing edge control surfaces and high lift devices because their hinge line is swept. To produce a reasonable CLMAX on a swept wing the hinge line of the inboard flaps may be made straight. Leading edge high lift devices are also used to improve the low speed characteristics.

SUPERCRITICAL AEROFOIL

A fairly recent design development, used to increase efficiency when operating in the transonic speed region, is the 'supercritical aerofoil'.



Because the airflow does not achieve the same increase of speed over the flattened upper surface compared to a conventional section, the formation of shock waves is delayed to a higher M_{FS} and are much smaller and weaker when they do form.

THE ADVANTAGES OF A SUPERCRITICAL AEROFOIL

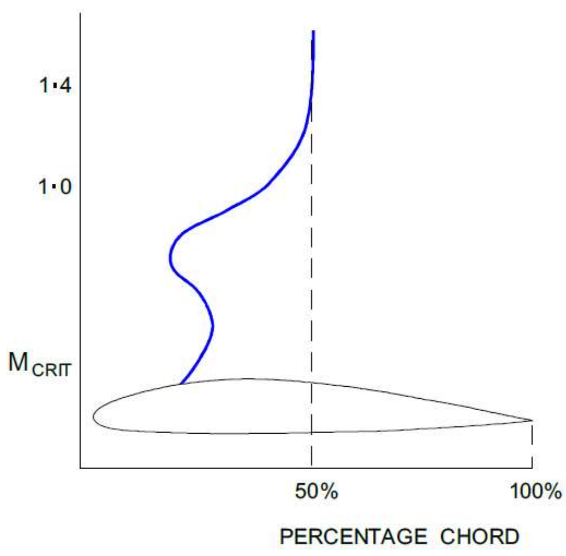
- Because of the delayed formation of shock waves and their weaker nature, less sweep angle is required for a given cruising Mach number, thus reducing some of the problems associated with sweepback.
- The greater thickness gives increased stiffness and strength for a given structural weight. This also allows a higher aspect ratio to be used which reduces induced drag.
- The increased section depth gives more storage space for fuel.

THE DISADVANTAGES OF A SUPERCRITICAL AEROFOIL

- The aerofoil front section has a negative camber to give optimum performance at cruise Mach numbers, but this is less than ideal for low speed flight. C_{LMAX} will be reduced, requiring extensive and complex high lift devices at the leading edge, which may include Krueger flaps, variable camber flaps, slats and slots.
- The trailing edge of the aerofoil has large positive camber to produce the 'aft loading' required, but which also gives large negative (nose down) pitching moments.
    - This must be balanced by the tailplane, causing trim drag and
    - Shock induced buffet may cause severe oscillations.

15) Effect of Shock wave on the Centre of Pressure

The centre of pressure of an aerofoil is determined by the pressure distribution around it. As the speed increases through the transonic region, the pressure distribution changes and the centre of pressure will move. It was shown that above M_{CRIT} the upper surface pressure continues to drop on the wing until the shock wave is reached. This means that a greater proportion of the 'suction' pressure will comes from the rear of the wing, and the centre of pressure is further aft. The rearward movement of the CP however is irregular, as the pressure distribution on the lower surface also changes. The shockwave on the lower surface usually forms at a higher free stream Mach number than the upper surface shock, but reaches the trailing edge first. As the aircraft accelerates to supersonic speed **the overall movement of the CP is aft to the 50% chord position.**



Rearward CP movement with increasing Mach number in the transonic region produces a nose down pitching moment.

This is known as '**Mach Tuck**', '**High Speed Tuck**' or '**Tuck under**'

A pull force become required. However, if shockwave occur ahead of the control surface, they become less effective. In addition, if the control is not deflected accordingly, the increase of the angle of attack of the wing will move the shock more aft and so increase in a nose down moment.

To resolve this the MACH TUCK or TUCK UNDER phenomena, a **MACH TRIMMER** is installed. This device sensitive to the Mach number may:

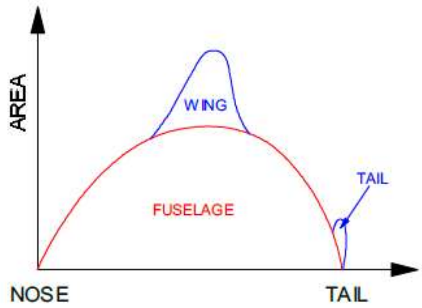
- Deflect the elevator up
- Decrease the incidence of the adjustable stabiliser
- Moves the CG rearward by transferring fuel from the wings to the rear trim tank

This ensure that the required stick force gradient is maintained in cruise at high Mach number.

16) Area Rule

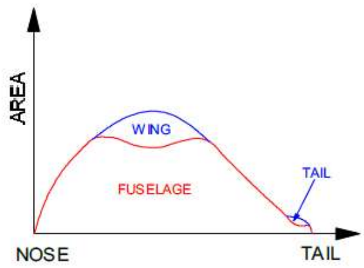
It was stated that in addition to the drag of individual components there is an extra drag due to interference between these components, principally between wing and fuselage.

This is especially important at high speed. Experiments have shown that a large part of the transonic drag rise for a complete aircraft is due to interference. Interference drag at transonic speeds may be minimised by ensuring that the cross-sectional area distribution along the aircraft's longitudinal axis follows a certain smooth pattern.



With some early high speed aircraft designs this was not the case. The area increased rapidly in the region of the wing, again in the vicinity of the tail and decreased elsewhere, giving an area distribution like the one illustrated above

On later aircraft, the fuselage was waisted, i.e., the area was reduced in the region of the wing attachment, and again near the tail, so that there was no "hump" in the area distribution, giving a distribution like the one illustrated below.



There is an optimum area distribution, and the minimisation of transonic interference drag requires that the aircraft should be designed to fit this distribution as closely as possible. This requirement is known as the **“area rule”**. In practice, no aircraft has this optimum distribution, but any reasonably smooth area distribution helps to reduce the transonic drag rise.

Old aircrafts which applied the area rule looked like a “Coke Bottle”. Flying a “Coke Bottle” mean the aircrafts is using the Area Rule design

17) Sonic Bang

The intensity of shock waves reduces with distance from the aircraft, but the pressure waves can be of sufficient magnitude to create a disturbance on the ground. Thus, "sonic bangs" are a consequence of supersonic flight. The pressure waves move with aircraft groundspeed over the earth surface.