DOCUMENT
GSM-G-CPL.004

DOCUMENT TITLE
AERODYNAMICS 3

HIGH SPEED FLIGHT

CHAPTER 2 – Transonic Flight

Version 1.0
February 2015

CONTENTS .... PAGE

SHOCK STALL

At airspeeds above the critical mach the formation of a shock wave and its associated pressure gradient results in a significant increase in drag and a reduction in lift.

This is caused by the sudden increase in pressure across the shock wave which causes localised heating of the air, and the eventual separation of the boundary layer behind the shock wave. At airspeeds just above the critical Mach number the increase in drag is mainly due to the loss of kinetic energy used in heating the air, which needs to be continuously supplied by the engines. With increasing Mach number the strength of the shock wave steadily increases, as does the size of the adverse pressure gradient, and this in turn determines the point at which the boundary layer separates from the surface. Both upper and lower surface shock waves can cause separation of the airflow and as in the case of low speed stall, the larger the adverse pressure gradient the larger is its associated turbulent wake. Refer to Figure 2.1.

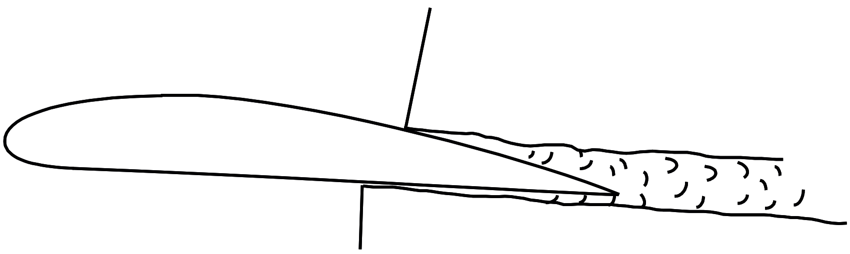

Figure 2-1 Aerofoil Separation Occurs behind the Shockwave

When separation firstly occurs, the coefficient of lift begins to fall and the coefficient of drag begins to rapidly rise. This phenomenon is known as the shock stall. It differs from a conventional low speed stall because it normally occurs at low angles of attack, although at high angles of attack the stall will occur at a lower mach number.

The combined effect of the energy loss across the ***turbulent wake*** behind the shock wave is known as ***wave drag***. Refer to Figure 2.2. and 2.3.

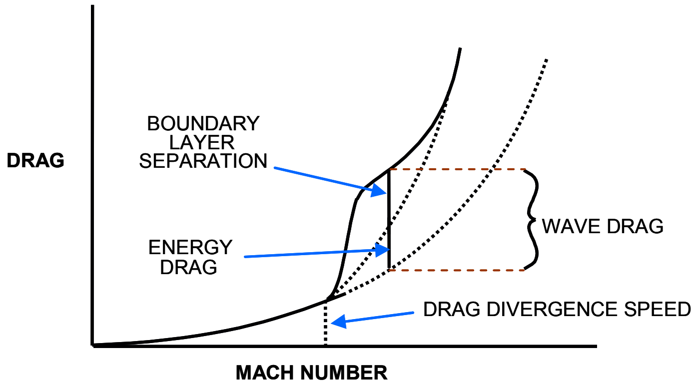

Figure 2-2 Transonic Wave Drag

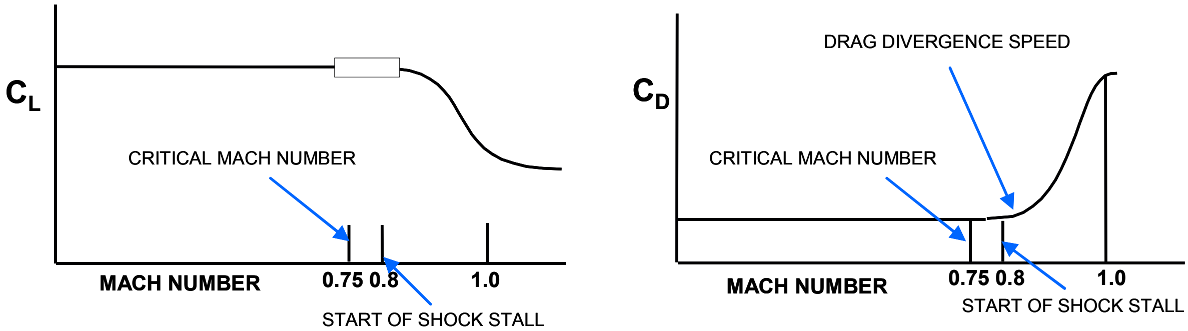

Figure 2-3 Effect of Shock and Stall on Lift and Drag

The drag varies significantly from the standard drag curve at the drag divergence speed, and the associated increase in drag is known as the transonic drag rise.

Similar to a high angle of attack low airspeed stall the separation of the boundary layer during the shock stall also results in buffeting of the aircraft, and a reduction in control effectiveness. So on high speed aircraft, the tail plane is always never in line with the wing so that the turbulent separated air misses the tail entirely. Refer to Figure 2.4.

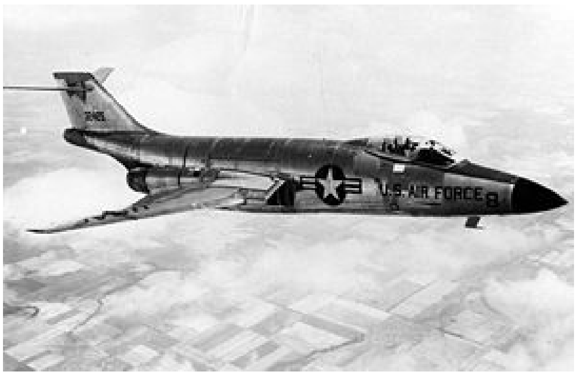

Figure 2-4 High Set Tail on a F101 Voodoo

Notably, as the upper layer surface shock wave moves rearwards with increasing Mach number, the region of shock induced separation reduces, and once the lower surface shock is established at the trailing edge some measure of recovery may occur.

THE EFFECT OF ALTITUDE ON SHOCK STALL

Apart from the engine limitations which affect an aircrafts maximum attainable altitude, it is also determined by the height at which an aircraft can fly without stalling

In practice, stalling can occur conventionally at low airspeeds and high angles of attack, or at high airspeeds due to shock stall. Accordingly, a specified range of flight speeds is attainable between the two limits at a given altitude. The margin between the two types of stall however narrows with increasing altitude. Notably the true airspeed of the low speed stall increases with altitude for a given indicated airspeed, whilst the true airspeed of the shock stall reduces up to the base of the stratosphere, and there after remains constant. Furthermore, during manoeuvres the two stalls will occur at considerably lower altitudes because the high angle of attack stalling speed increases while the shock stalling speed decreases.

INCREASING M_{CRIT} AND DELAYING THE TRANSONIC DRAG RISE

Because of the undesirable effects associated with shock waves various design features are used to either overcome or reduce these effects. In subsonic jet aircraft this is achieved by using designs which increase the value of M_{crit}, so that an aircraft enters the transonic range at a higher airspeed and passes through the transonic range faster.

Other aircraft incorporate features which are designed not to delay the onset of transonic flow, but instead to minimise the adverse effects. This is mainly accomplished by various design features which affect the wing and tail plane of an aircraft.

WING SECTION

The following design features of a wing section are used to increase M_{crit}:

- Low thickness/chord ratio
- Maximum thickness well aft
- Small leading edge radius of curvature.

A thinner wing section will increase M_{crit} by giving less of an increase in speed over the wing compared to a thicker wing.

But thin wings are not good structurally. They need to be of low aspect ratio for sufficient strength and have a sharp leading edge for low transonic drag. Thin wings also cannot house undercarriage or fuel so are limited in their use. They also have very high stalling and landing speeds which are also great disadvantages. So a better option is to sweep the wing back.

SWEEP BACK

Figure 2-6 35° Sweepback on a Sabre

When a wing is swept back the effective chord, (parallel to the aircrafts longitudinal axis), is lengthened, but the wing's thickness remains unchanged. Refer to Figure 2.6

In effect, the thickness/chord ratio is therefore reduced, which in turn results in a higher value of M_{crit}, and a delayed transonic drag rise. Refer to Figure 2.7.

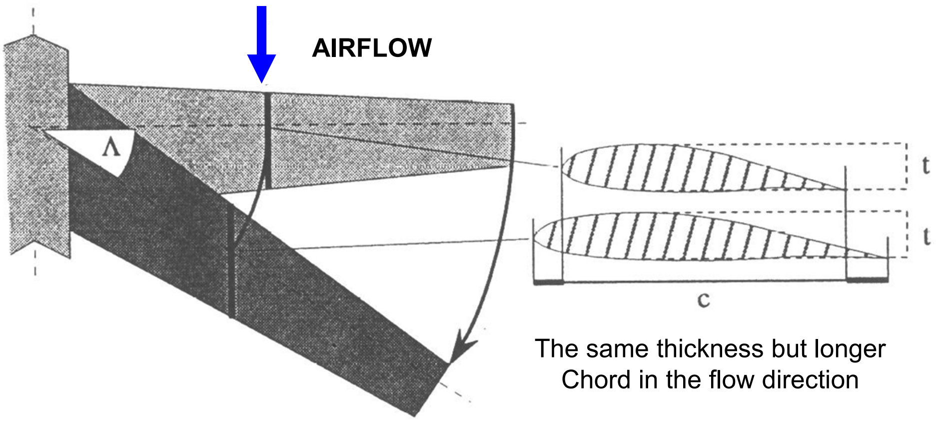

Figure 2-7 Effect of Sweepback on Aspect Ratio

It follows that the greater the sweepback the higher the value of M_{crit}, and the greater the reduction in drag under all transonic speeds.

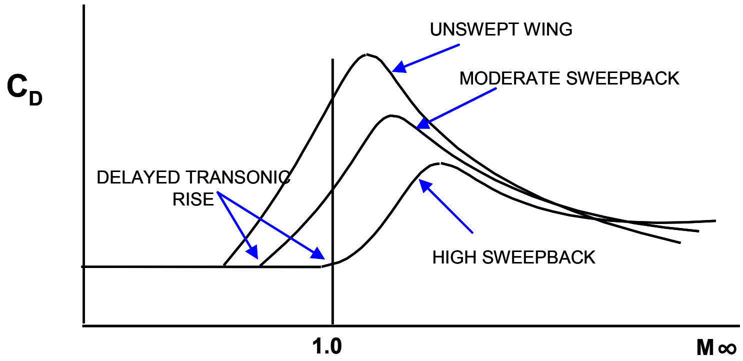

Figure 2-8 Effect of Sweepback on Transonic Drag Rise

Sweepback advantages: A few of the less significant advantages of sweepback are as follows:

1. ***The wing lift curve slope*** is reduced for a given aspect ratio. This is illustrated by the lift curve comparison for the straight and swept wing. Any reduction of lift curve slope implies the wing is less sensitive to changes in angle of attack. This is a beneficial effect only when the effect of gusts and turbulence is considered. Since the swept wing has the lower lift curve slope it will be less sensitive to gusts and experience less "bump" due to gust for a given aspect ratio and wing loading. This is a consideration particular to the aircraft whose structural design shows a predominating effect of the gust load spectrum, for example, transport, cargo, and patrol types.

2. ***"Divergence***" of a surface is an aeroelastic problem which can occur at high dynamic pressures. Combined bending and twisting deflections interact with aerodynamic forces to produce sudden failure of the surface at high speeds. Sweep forward will aggravate this situation by "leading" the wing into the airstream and tends to lower the divergence speed. On the other hand, sweepback tends to stabilize the surface by "trailing" and tends to raise the divergence speed. By this tendency, sweepback may be beneficial in preventing divergence within the anticipated speed range. Refer to Figure 2.8.

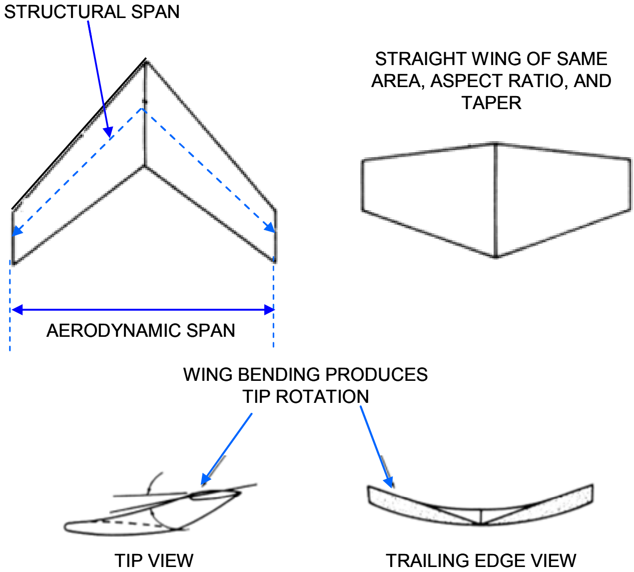

Figure 2-9 Structural Complications due to Sweepback

3. ***Sweepback*** contributes slightly to the static directional—or weathercock—stability of an aircraft. The wing into the wind has less sweep and a slight increase in drag. The wing away from the wind has more sweep and less drag. The net effect of these force changes is to produce a yawing moment tending to return the nose into the relative wind. This directional stability contribution is usually small and of importance in tailless aircraft only.

4. ***Sweepback*** contributes to lateral stability in the same sense as dihedral. When the swept wing aircraft is placed in a sideslip, the wing into the wind experiences an increase in lift since the sweep is less and the wing away from the wind produces less lift since the sweep is greater. The swept wing aircraft in a sideslip experiences lift changes and a subsequent rolling moment which tends to right the aircraft. This lateral stability contribution depends on the sweepback and the lift coefficient of the wing. A highly swept wing operating at high lift coefficient usually experiences such an excess of this lateral stability contribution that adequate controllability may be a significant problem.

Sweepback disadvantages: Although sweepback is a great asset in increasing the critical mach number it does have a number of disadvantages, which are namely:

- A reduction in the coefficient of lift, which increases stall speed
- Wing tip stalling leading to pitch-up
- Low aspect ratio leading to increased induced drag at high angles of attack, which is particularly dangerous during take offs and landings.

When sweepback is combined with taper there is an extremely powerful tendency for the wing to stall tip first. This pattern of stall is very undesirable since there would be little stall warning, a serious reduction in lateral control effectiveness, and the forward shift of the centre of pressure would contribute to a **nose-up moment** ("pitch-up" or "stick force lightening"). Taper has its own effect of producing higher local lift coefficients toward the tip and one of the effects of sweepback is very similar. All outboard wing sections are affected by the upwash of the preceding inboard sections and the lift distribution resulting from sweepback alone is similar to that of high taper.

An additional effect is the tendency to develop a strong spanwise flow of the boundary layer toward the tip when the wing is at high lift coefficients. This spanwise flow produces a relatively low energy boundary layer near the tip which can be easily separated. The combined effect of taper and sweep present a considerable problem of tip stall .Design for high speed performance may dictate high sweepback, while structural efficiency may demand a highly tapered planform. When such is the case, the wing may require extensive aerodynamic tailoring to provide a suitable stall pattern and a lift distribution at cruise condition which reduces drag due to lift. Washout of the tip, variation of section camber throughout span, flow fences, slats, leading edge extension, etc., are typical devices used to modify the stall pattern and minimize drag due to lift at cruise condition.

(2) The use of sweepback will reduce the lift curve slope and the subsonic maximum lift coefficient. It is important to note this case is definitely subsonic since sweepback may be used to improve the transonic manoeuvring capability. Various sweep angles applied to wings of moderate aspect ratio produce these approximate effects on the subsonic lift characteristics:

(3) The use of sweepback will reduce the effectiveness of trailing edge control surfaces and high lift devices. A typical example of this effect is the application of a single slotted flap over the inboard 60 percent span to both a straight wing and a wing with 35° sweepback. The flap applied to the straight wing produces an increase in maximum lift coefficient of approximately 50 percent. The same type flap applied to the swept wing produces an increase in maximum lift coefficient of approximately 20 percent. To produce some reasonable maximum lift coefficient on a swept wing may require un-sweeping the flap hinge line, application of leading edge high lift devices such as slots or slats, and possibly boundary layer control.

(4) As described previously, sweepback contributes to lateral stability by producing stable rolling moments with sideslip. The lateral stability contribution of sweepback varies with the amount of wing sweepback and wing lift coefficient—large sweepback and high lift coefficients producing large contribution to lateral stability. While stability is desirable, any excess of stability will reduce controllability. For the majority of airplane configurations, high lateral stability is neither necessary nor desirable, but adequate control in roll is absolutely necessary for good flying qualities. An excess of lateral stability from sweepback can aggravate "Dutch roll" problems and produce marginal control during crosswind takeoff and landing where the aircraft must move in a controlled sideslip. Therefore, it is not unusual to find swept wing aircraft with negative dihedral and lateral control devices designed principally to meet crosswind takeoff and landing requirements.

SUPERCRITICAL WINGS

Supercritical wings were developed by Dr Richard Whitcomb in the early 70's at NASA to help reduce the severity of the shock stall and allow aircraft to travel faster. Most modern jet transport category aircraft since 1975 are fitted with supercritical wings. Refer to Figure 2-10.

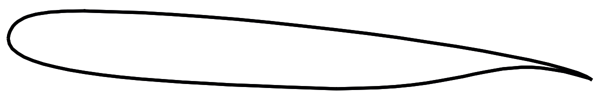

Figure 2-10 Supercritical Wing Section

In this design the wing is flatter on top to delay shock wave formation and the upper trailing edge is curved (reflex) to restore the lift lost by flattening the top of the wing. This also allows us to have a wing that is slightly thicker than a conventional section which means more strength and internal fuel for 5% less drag. This equates to roughly a 2.5% increase in operating profits for an airliner.

The flow is then gradually decelerated near the trailing edge to a subsonic speed to discourage the formation of shock waves. (Fig 2-11)

The only real disadvantage of the supercritical wing is the reflex near the trailing edge can aggravate Mach tuck due to the centre of pressure being further back.

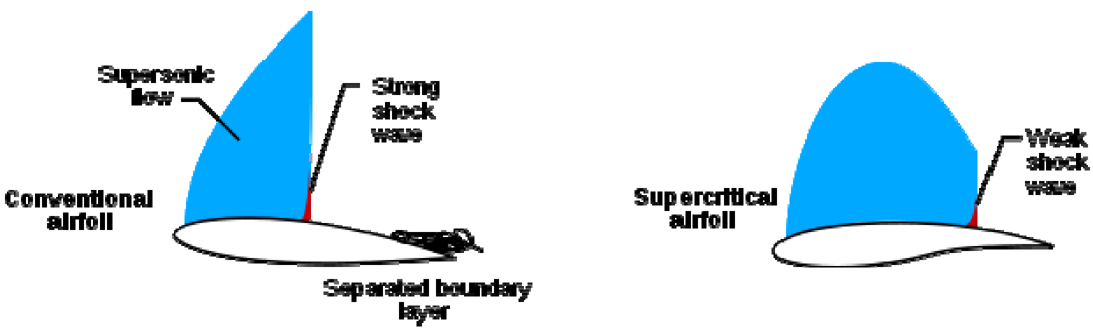

Figure 2-11 Conventional v Supercritical Wing

Figure 2-12 Supercritical Wing note the upside down appearance of the supercritical wing

VORTEX GENERATORS

These are small wing like surfaces that project vertically into the airstream They operate by forcing high energy air into the boundary layer thereby enabling it to overcome the adverse pressure gradient caused by the shock wave, and thus delaying its separation. Refer to Figures 2-13 and 2-14.

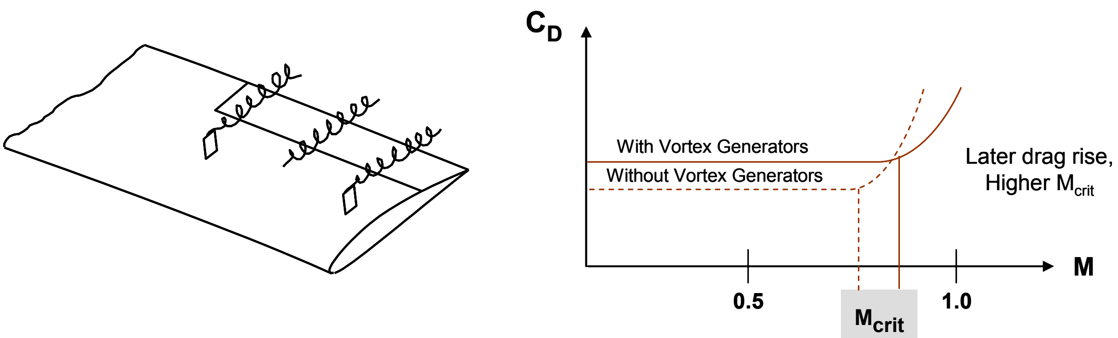

Figure 2-12 Vortex Generator Performance

Figure 2-13 Vortex Generators on an A4 Skyhawk

They also generate a weak set of shock (expansion) waves and this forces the supersonic air to follow the contour of the wing with much less separation. The shock wave is further back and reaches the trailing edge sooner, so M_{crit} is increased. This is shown in the following picture. (Fig 2-14) The Vortex generators can be seen on the upper aerofoil.

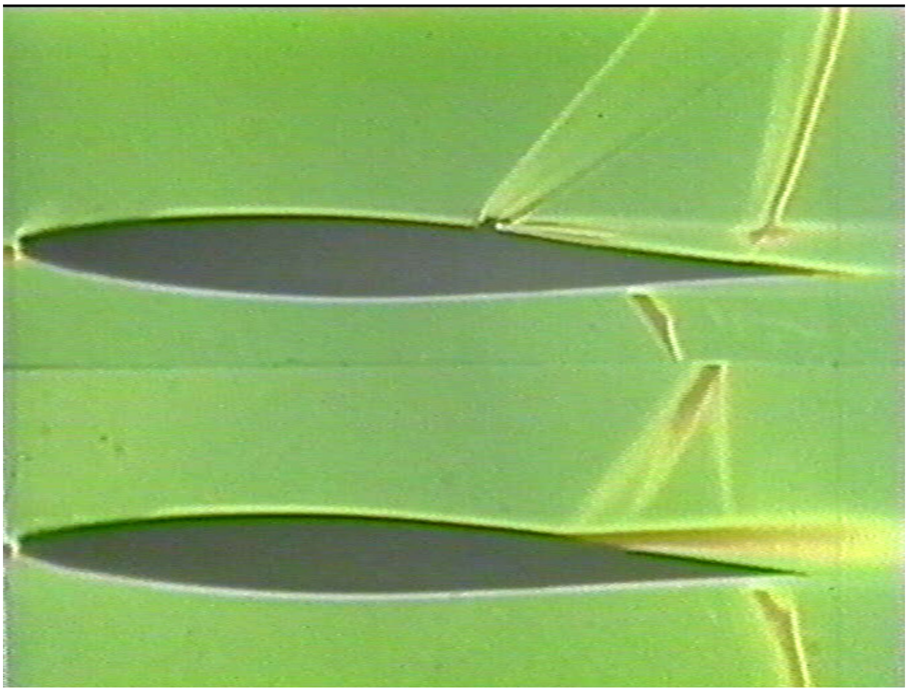

Figure 2-14 Effect of vortex generator on shock wave

EFFECT OF WEIGHT AND LOAD FACTOR ON M_{CRIT}

For a given EAS / Mach number, **an increase in aircraft mass** requires an increase in lift coefficient, and therefore **an increase in angle of attack.**

Greater acceleration over the upper wing surface means earlier sonic flow and therefore **lower M_{crit}.**

An **increase in load factor** caused by (for example) a steep level turn also requires higher lift coefficient and angle of attack: therefore greater relative airflow acceleration and **lower M_{crit}** Refer to Figure 2-15.

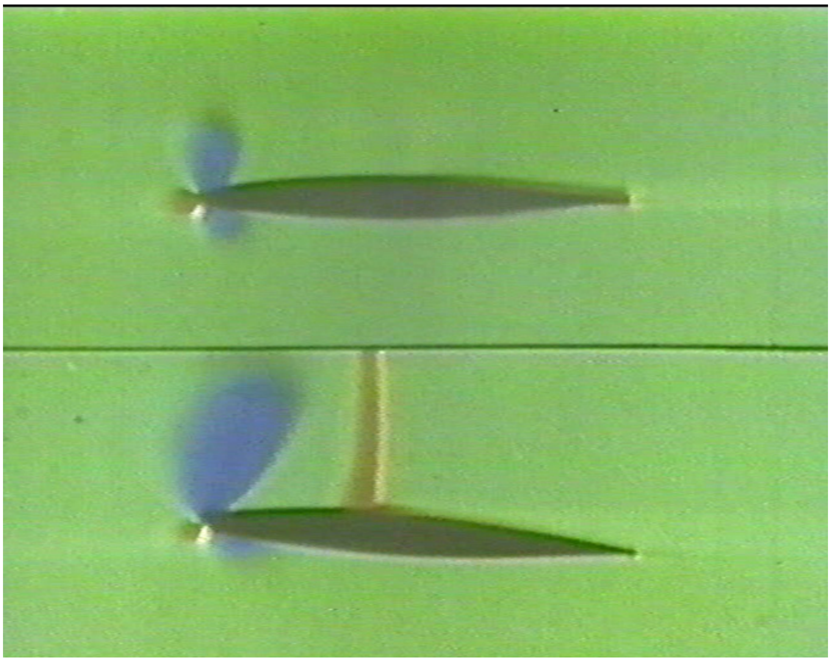

Figure 2-14 two identical aerofoils at same Mach number but different angles of attack, showing the lower aerofoil has a lower M_{crit} due to the increased angle of attack.

AREA RULE

The design feature used for mating the various parts of the aircraft together to produce minimum drag is called Area Rule. On theoretical grounds it can be shown that there is a certain ideal shape for a body of specified volume or frontal area which has minimum drag at a specified transonic or supersonic speed. A graph distribution of its cross-sectional area along its longitudinal axis is a smooth curve. The area rule states that, if an irregular body such as an aircraft has the same cross-sectional area distribution as this ideal body, it will also have the minimum possible drag. Figure 2-16 illustrates this principle.

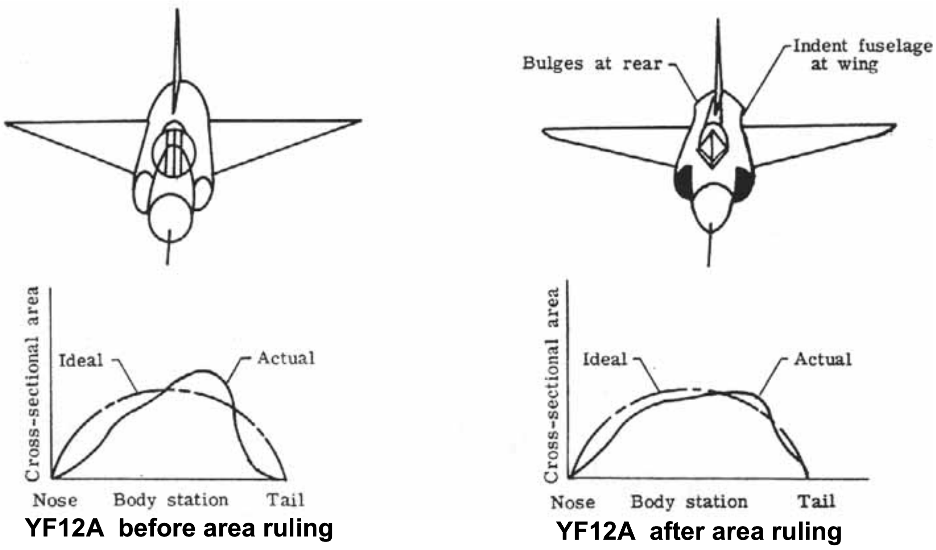

Figure 2-16 Area Ruling

The presence of a lump corresponding to the aircrafts wings will obviously mean high velocities round it and a very strong shockwave behind it. If the area is increased as shown by the dotted line as shown in Figure 2.16, while keeping the same maximum thickness, then the acceleration will be decreased with a corresponding decrease in drag.

Area ruling can be achieved in several ways:

1. by "waisting" the fuselage
2. by "thickening" the fuselage ahead of and behind the wings as shown in Figure 2-17
3. by combining a flat-sided fuselage with highly swept wings.

Figure 2-17 Area Ruling on a F102 Delta Dart

CONTROL PROBLEMS IN TRANSONIC FLIGHT

When shock waves form on an aircraft's wing the resulting shock induced separation can lead to vibration and control surface ineffectiveness Refer to Figure 2-18.

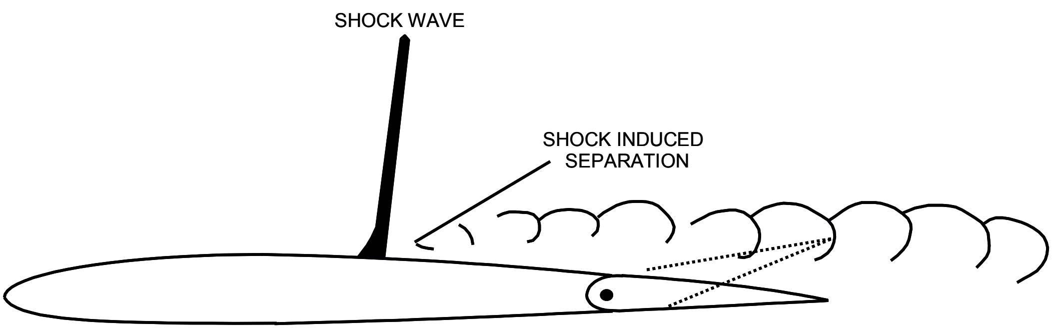

Figure 2-18 Shock Induced Separation

The disturbed air over the control surfaces may even cause un-commanded erratic movements. In Figure 2-19, we can see that the aileron is not fully effective because any changes behind the shock wave can't affect the air in front of it. The aileron is then in separated and turbulent airflow so has lost effectiveness.

This accordingly alters the pressure distribution over the front of the wing, thereby changing the position of the centre of pressure and the wings overall pitching moment. This in turn alters the wings angle of attack and results in rapid movements, backwards and forwards of the shock waves Refer to Figures 2-19 and 2-20.

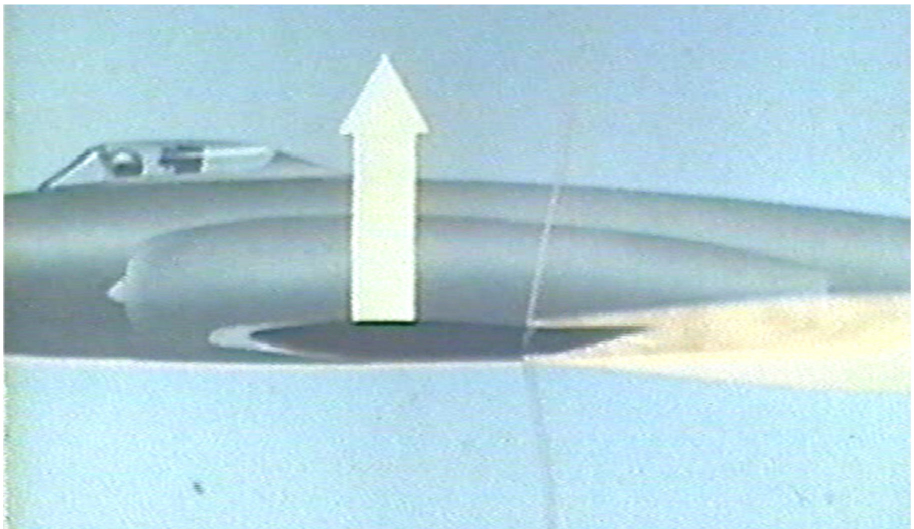

Figure 2-19 Shock Induced Separation

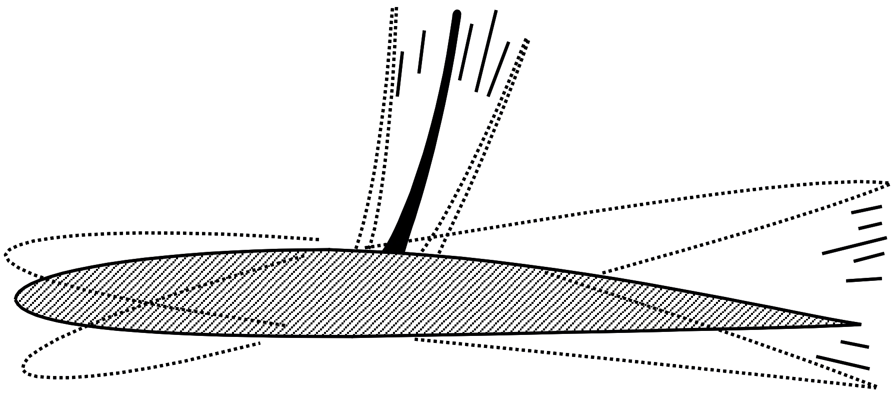

Figure 2-20 Effect of shock waves on pitching moment

This ultimately sets-up a kind of instability (**flexural flutter**) and the associated rapid changes in the pressure distribution result in vibration of the whole aircraft.

If shock waves form on a control surface it will also affect the stick forces by altering the hinge moment Refer to figure 2-21.

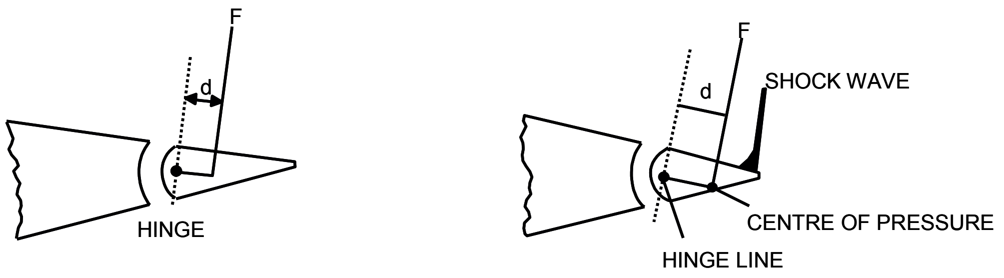

Figure 2-21 Variation of hinge moment with shockwave movement

Since the hinge moment opposing the movement of the control surfaces is a product of the force acting through its centre of pressure times its distance from the hinge line, it will fluctuate, as will the stick force in phase with any shock wave movement. It follows that rearward movement of a shock wave acting on a control surface will result in increasing stick forces, reaching a maximum value with the shock wave at its trailing edge. Refer to Figure 2-22.

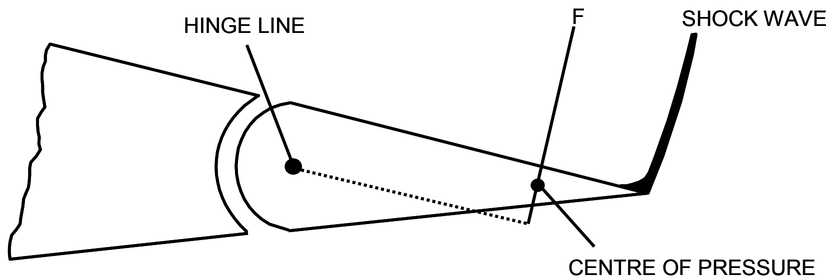

Figure 2-22 Hinge moment reaches maximum when shock wave reaches trailing edge of control surface

Furthermore if the centre of pressure moves ahead of the hinge-line transitory overbalance will occur and control surface reversibility will take place. Refer to Figure2-23.

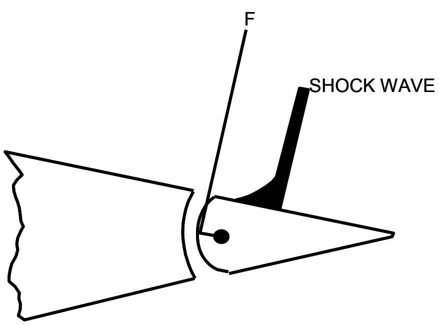

Figure 2-23 Centre of pressure ahead of the hinge line causes transitory overbalance

Since the shock waves move quickly with changes in control surface deflection the effect is felt on the flight deck as snatching or buffeting depending on the position of the control surface. The disturbed air resulting from shock induced separation also precludes the use of aerodynamic balance methods, in particular tabs. **Power operated controls are almost essential in preference to manually operated controls.**

Other methods used to overcome these control problems are:

- All moving surfaces
- Surfaces of low thickness/chord ratio
- Larger control surfaces.
- Position of the tailplane
- Fitment of vortex generators.

ALL MOVING TAILPLANE

By having an all moving tailplane the larger surface does not have to move as far to achieve the same result, so therefore shockwaves have no real effect as there is no hinge line as with a separate tailplane and elevator. They can also be very thin (high thickness /chord ratio) which gives low drag and high. M_{crit}. Refer to figure 2-24.

Figure 2-24 F16 All Flying Tail Plane

VORTEX GENERATORS

As described previously in this text, Vortex generators also help control effectiveness at high as well as low speed. At low speed, they re-energise a sluggish boundary layer by using the vortex generated to grab high energy free stream airflow and pull it down into the separated boundary layer and therefore keeping the control effective.

As described, at high speed they generate a weak expansion shock wave which delays separation and raises M_{crit}.

Figure 2-25 A4 vortex generators

Vortex generators fitted to the wing of an A4 Skyhawk, are in front of the aileron to ensure positive control at low as well as high speed. Refer to Figure 2-25.

TRANSONIC EFFECT ON AIRCRAFT TRIM AND STABILITY

In transonic flight it is the nature of the airflow behind a shock wave which determines the stability characteristics of an aircraft.

Longitudinal Stability: This is the most commonly affected form of stability. It is determined by movement of the centre of pressure, coupled with changes in the coefficient of lift, and the amount of downwash acting on the tailplane. In straight and level flight at low subsonic airspeeds the centre of pressure is typically located aft of the centre of gravity Refer to Figure 2-26.

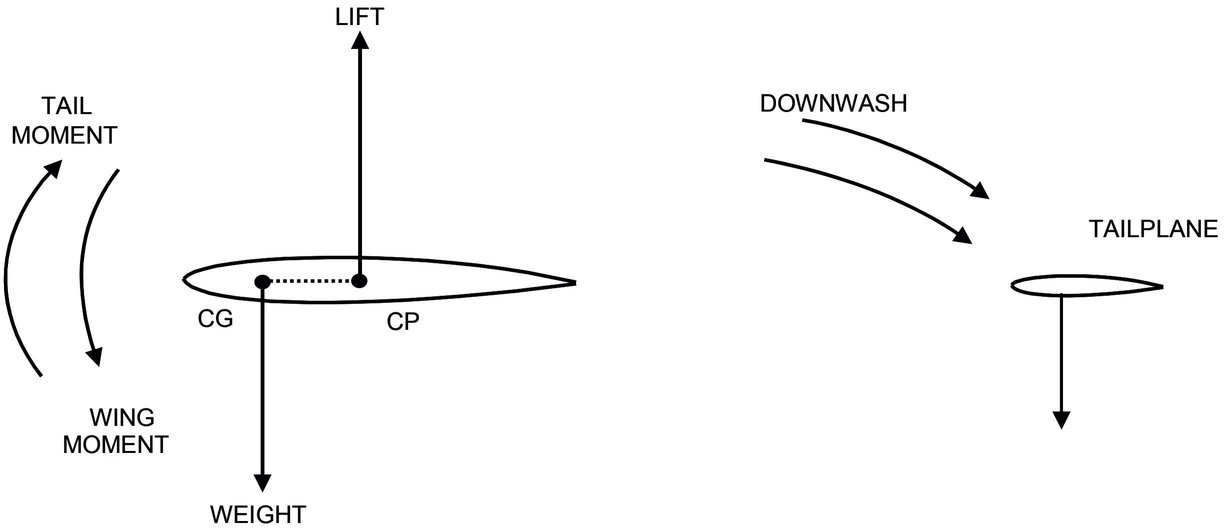

Figure 2-26 Wing moment balanced by tail moment

This results in a nose-down pitching moment which is counteracted by placing a small download on the tailplane. When the shock stall takes place, lift aft of the shock wave is destroyed and the tailplane becomes covered in disturbed airflow. Refer to Figure 2-26.

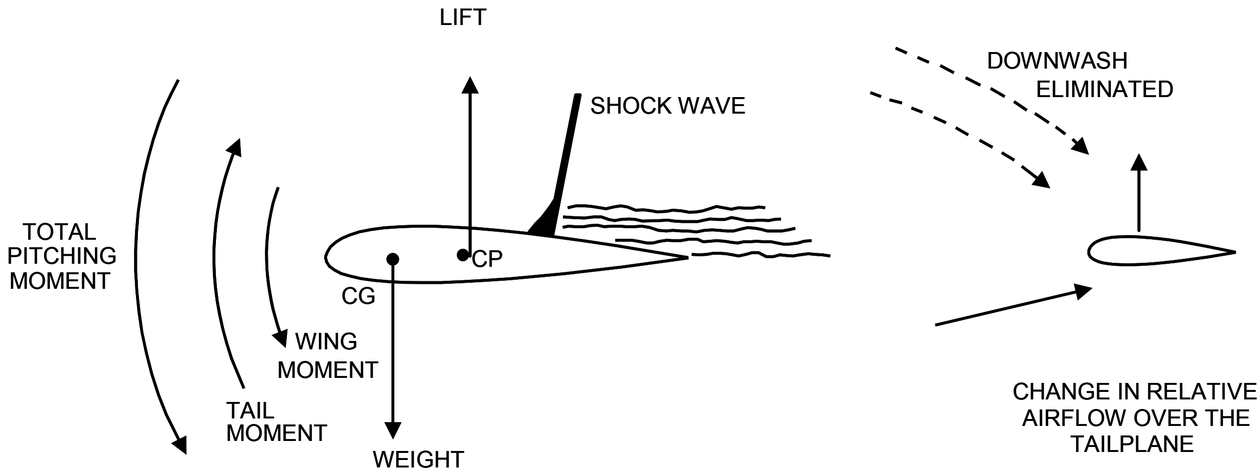

Figure 2-27 Shock induced separated airflow over the tail plane produces Mach tuck

The downwash acting on the tailplane is consequently eliminated causing a smooth and radical change in its angle of attack. The angle of attack becomes more negative and the download acting on the tailplane becomes an upload, which in conjunction with the wing pitching moment causes a violent nose-down pitching moment, normally referred to as **Mach tuck.**

The exact nature and strength of the changes in trim and stability is however dependent on the design of an aircraft.

Lateral Stability: Disturbances about the longitudinal axis are often encountered in transonic flight and are characterised as a wing heavy tendency as the critical Mach number is exceeded. This occurs because shock waves do not always form simultaneously, or at identical places on opposite wings.

The design features which normally provide lateral stability may consequently reverse the effect and aggravate a dropping wing. This occurs because the downgoing wing sideslips causing the airflow to accelerate and intensify the shock wave, causing further wing drop.

Directional Stability: This form of stability is similarly affected by the variation in wing shock wave formations, which result in different drag characteristics. For example if a shock wave first forms on the left wing, the associated increase in drag will cause the aircraft to yaw in the same direction. While yawing to the left the airflow will then accelerate over the right wing, so intensifying the shock wave and increasing the drag. This process is thus self perpetuating and results in snaking or Dutch roll, depending on the lateral and directional characteristics of an aircraft.

Mach Trim: To guard against **Mach tuck** which can occur in transonic flight frequent pitch trim changes are required. To facilitate this, a variable incidence tailplane is fitted, which is automatically positioned by way of a mach trim system. This system is designed to aid aircraft longitudinal stability and ensures that the forward stick forces increase proportionally with increasing Mach number.

It controls the longitudinal trim system and will only have an input when the pilot is not manually trimming or the auto pilot is not engaged. Refer to Figure 2-28.

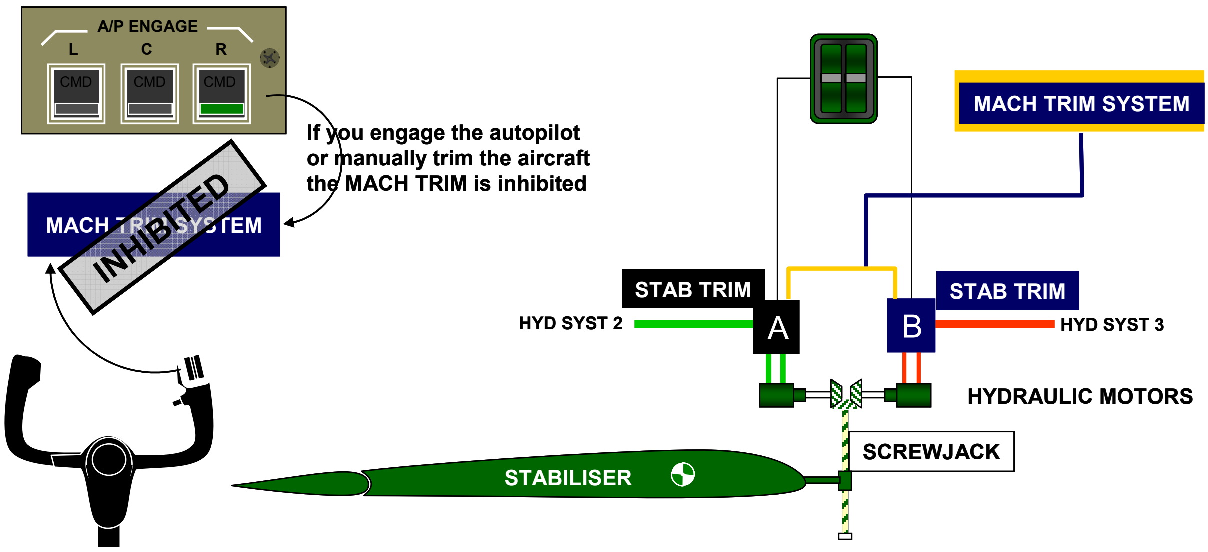

Figure 2-28 Mach Trim System