PART 1 : SECTION 1

CHAPTER 14

TRANSONIC AND SUPERSONIC AERODYNAMICS

Introduction

1. Low-speed aerodynamics are based largely on the assumption that air is an incompressible fluid ; the attendant errors are negligible since at low speeds the amount of compression that occurs owing to the passage of an aircraft is also negligible. However, at flight speeds approaching that of sound, at which pressures about the aircraft are becoming large, the compressible nature of air makes itself evident in a number of ways which have a marked effect on aircraft in flight. A knowledge of the fundamental changes in the airflow brought about by the compressibility of air and their effects on the aircraft enables a pilot to anticipate and understand certain peculiarities in aircraft behaviour.

2. Whereas at subsonic speeds a certain type of flow pattern is established by the air moving around the wing, at supersonic speeds an entirely different type of pattern exists. In pure subsonic or supersonic flight the airflow around a given wing can be controlled and its behaviour predicted with some accuracy, but in the transition (transonic) period, marked control and stability difficulties arise unless the aircraft has certain design features which minimize these effects. The following is a simplified explanation of the fundamental changes experienced in the airflow at high speeds.

Air Pressure

3. At standard sea-level temperature individual molecules of air are in constant motion in random directions at a speed of about 1,700 f.p.s. (1,000 knots). Air pressure is thus the total effect of the impact of air molecules on any surface exposed to their movements. The speed of the molecules depends on their temperature, the higher the temperature the greater the speed. Thus a small balloon which is warmed expands as the molecular speed rises with temperature and so increases the pressure.

Pressure Disturbances and Sound Waves

4. The speed of propagation of any pressure disturbance is closely connected with the speed at which the molecules are moving. Since the molecules move in a random manner and are constantly colliding with each other and rebounding, a pressure disturbance moving in a fixed direction will travel at a lower speed than that of individual molecules. This lower speed (about 1,118 f.p.s. or 662 kts. at standard sea-level temperature) is the speed at which small pressure disturbances travel through the air. Since sound waves are small pressure disturbances, this speed is also the speed of sound. If the balloon (see para. 3) were to burst, the sound waves would travel at 1,118 f.p.s., but the speed of the released air would be only 4 or 5 f.p.s.

5. **Mach Number.** Pressure waves and their rate of movement have a profound effect on high-speed (transonic and supersonic) flight. It is therefore necessary to compare the T.A.S. of an aircraft to the speed of sound at the height, *i.e.* temperature, at which the aircraft is flying. This comparison is expressed by the *mach number* (M) indicated by the machmeter on the instrument panel. The mach number is the value of the T.A.S. of the aircraft divided by the local speed of sound (a), *i.e.* $M = \frac{V}{a}$.

6. A stationary object which vibrates at a certain frequency is the source of a continuous series of pulses of compressed air. These small disturbances, corresponding to the ripples produced when a stone is dropped into water, move out as expanding spheres travelling at the speed of sound (Fig. 1(a)). If the source of disturbance starts moving, it closes up on the pressure waves ahead of it (Fig. 1(b)). Thus an observer standing ahead of the object would receive more sound waves per second (a higher frequency) than one standing behind. The ears interpret this higher frequency as a higher pitched note which drops to a lower note after the object passes—this is the well-known Doppler effect which is evident whenever a low-flying aircraft approaches rapidly and passes overhead.

7. **Mach Cone.** If the object were moving at the speed of sound, the pressure disturbances would accumulate immediately in front in the form of a continuous line or wave (Fig. 1(c)). At greater speeds (Fig. 1(d)) the object would be travelling faster than the pressure disturbances.

An infinite number of pressure waves would produce a continuous line, inclined backwards in the form of a cone. The angle of the cone, called the *mach angle*, would become smaller as the speed rose. This cone, which marks the boundary of the sphere of influence of the body, is called a *mach cone*. All objects within the mach cone would experience the effects of the passage of the body, all those outside would be unaffected.

8. At subsonic speeds the pressure waves that travel ahead serve, in effect, to warn the air ahead of the approach of an object, enabling individual particles of air to adjust their position in readiness for the passage of the object. This effect is clearly illustrated by streamlines about a wing at subsonic speed ; at some distance ahead of this wing the airflow is already changing direction with respect to the free-stream velocity.

9. In supersonic flight, air particles ahead of the mach cone have no warning of the approach of the object until they are violently deflected from their state of equilibrium. This violent displacement of air leads to a considerable increase in drag ; it is considered in greater detail in subsequent paragraphs.

Transonic Airflow

10. If an aerofoil is placed into a free subsonic airstream moving at about ·75M, the aerofoil has the usual subsonic accelerating effect on the air which is moving towards the point of maximum thickness. If the free-stream speed is V, and the increase in speed is v then the peak speed reached at the point of maximum thickness is V + v (Fig. 2).

11. **Critical Mach Number.** If the free-stream speed is then increased, the peak speed follows suit. It is apparent that the speed of sound must first be reached over the point of maximum thickness. The free-stream speed at which sonic speed is first reached over the wing is the critical velocity (V crit) for the particular aerofoil section. It can be seen that V crit + v = a and from this, that V crit = $a - \text{v}$. The critical speed is therefore always lower than the speed of sound. The free-stream mach number prevailing when V crit is reached is termed the critical mach number (M crit). This mach number is of great significance in flying for range since it marks the beginning of the compressibility drag rise. *The critical mach number is defined as the mach number at which, because of compressibility effects, the drag coefficient of an aircraft begins to increase.*

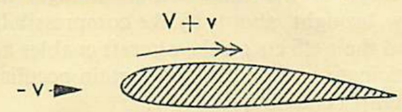

Fig. 2. Effect of Aerofoil on Free Airflow

12. At speeds below V crit the airflow over the entire wing is subsonic and therefore pressure differences at any point can travel forward to be communicated to the oncoming airflow which can then adjust its path accordingly, tending always to flow towards low-pressure areas.

13. **Shock Waves.** At V crit the air that is being accelerated can no longer be affected by conditions aft of the point of maximum thickness, since the pressure disturbances, which travel at the

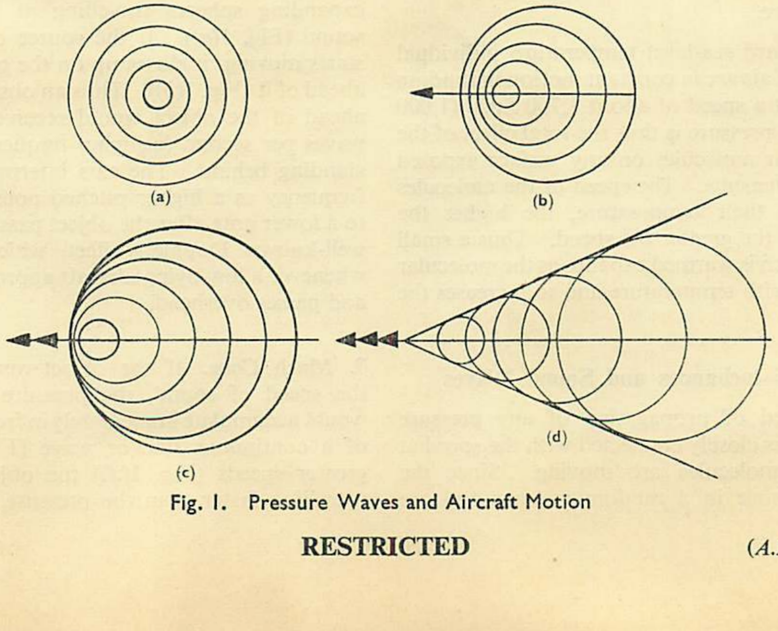

Fig. 1. Pressure Waves and Aircraft Motion

speed of sound, cannot move forward beyond the point at which sonic speed exists. Pressure waves from the rear (subsonic) portion of the wing therefore accumulate immediately behind the sonic area until a sound wave of finite amplitude is formed. This is termed a *shock wave*, or shock, and at a constant free-stream mach number it remains stationary on the wing.

14. The pressure on the subsonic side, *i.e.* aft of the shock wave, may be two or three times greater than the pressure on the other side. This marked incease in pressure occurs within the thickness of the shock wave, which is about one ten-thousandth of an inch. The sudden drop from sonic, or slightly supersonic, speed to subsonic speed is also concentrated within the shock wave and is thus practically instantaneous. Some of the kinetic energy lost in this violent deceleration is dissipated as heat, but most provides the pressure rise behind the wave.

15. It should be noted that at no time is there any "piling up of air" in front of or on the wing—this oft-quoted misconception of compressibility and a shock wave is completely erroneous ; the air at all times flows steadily over the wing subject to the accelerations and decelerations imposed by the aerofoil shape and attendant shock waves.

16. Increase of the free-stream speed to above M crit progressively enlarges the area of wing covered by supersonic flow, the area growing both in a forward and backward direction. As the supersonic area moves aft, the shock wave retreats towards the trailing edge and finally reaches it, to which point it remains attached during the period of supersonic flight. At this stage the entire wing is covered by supersonic flow and free-stream speed has reached mach 1.

17. However, a small subsonic region still exists immediately in front of a rounded leading edge. Pressure impulses from such a leading edge move forward slightly to a point where the air has slowed down to sonic speed as a result of the cushioning effect of the leading edge. Here an accumulation of these impulses results in the formation of a second shock (bow) wave. Fig. 3 shows, in simplified form, the transition between subsonic and supersonic flow—the transonic stage. It shows also the essential difference between subsonic and supersonic airflow. The effects described above are duplicated over the lower surface of the wing but, as the wing is at an angle of attack, they are usually not quite so marked.

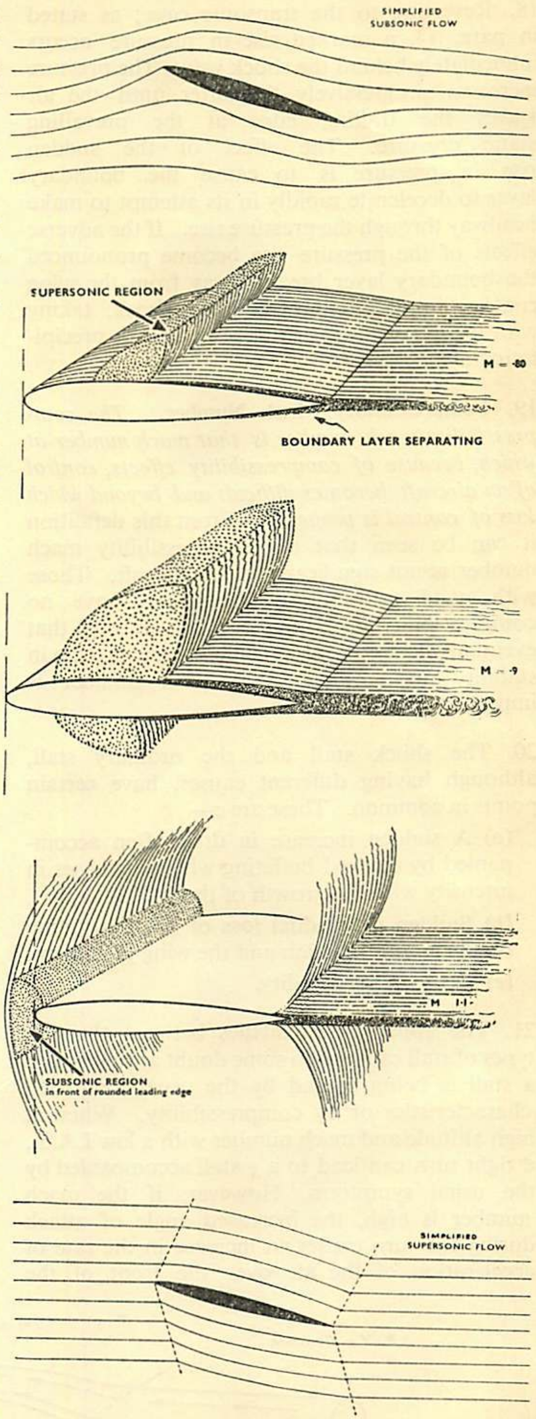

Fig. 3.
Shock Wave Development at Increasing Mach Number

Shock Stalls

18. Reverting to the transonic case; as stated in para. 14, a marked rise in pressure occurs immediately behind the shock wave. The pressure increases progressively thereafter until the air leaves the trailing edge at the prevailing static pressure. The effect of the sudden rise in pressure is to cause the boundary layer to decelerate rapidly in its attempt to make headway through the pressure rise. If the adverse effects of the pressure rise become pronounced the boundary layer breaks away from the wing contour immediately behind the shock, taking with it the layers of air above it, and so precipitating a shock stall.

19. **Compressibility Mach Number.** *The compressibility mach number is that mach number at which, because of compressibility effects, control of an aircraft becomes difficult and beyond which loss of control is probable.* From this definition it can be seen that the compressibility mach number is not significant on all aircraft. Those with good transonic characteristics have no compressibility mach number; but on those that eventually lose control, or suffer a serious drop in stability and control, this mach number is important.

20. The shock stall and the ordinary stall, although having different causes, have certain points in common. These are :—

(*a*) A sudden increase in drag often accompanied by marked buffeting which increases in intensity with the growth of the stall.

(*b*) Sudden or gradual loss of lift, depending on the aerofoil section and the wing plan form.

(*c*) Decrease of stability.

21. The apparent similarities between the two types of stall can lead to some doubt as to whether a stall is being caused by the usual low-speed characteristics or by compressibility. When at high altitude and mach number with a low I.A.S., a tight turn can lead to a *g* stall accompanied by the usual symptoms. However, if the mach number is high, the increased angle of attack during the turn causes an increase in the rate of acceleration of the air over the front of the wing. This may bring on compressibility effects and a shock stall at a mach number lower than the compressibility mach number, the symptoms being the same as those of the *g* stall. Since the basic cause of both conditions is a high angle of attack, recovery is made by decreasing the *g*.

Difference Between Subsonic and Supersonic Flow

22. In any convergent channel (the narrowing portion of a venturi) a subsonic flow of air, when compared to the free stream, is accelerated ; and the static pressure decreases. In an expanding channel (the tail of a venturi) the air decelerates and the static pressure rises.

23. *In a supersonic free stream the converse applies.* This is an important phenomenon, and unless it is appreciated the characteristics of supersonic airflow are not easily understood. The physical differences between aircraft designed for subsonic and those designed for supersonic flight are attributable entirely to this single reason. The table below sets out the basic differences between the two types of flow.

	Subsonic Flow	Supersonic Flow
Contracting Channel	Accelerates and static pressure decreases.	Decelerates and static pressure increases.
Expanding Channel	Decelerates and static pressure increases.	Accelerates and static pressure decreases.

24. It is an established physical law that when the pressure on a given volume of air is reduced, then that air *expands*. It is also known that when a subsonic airflow is passing over a wing, the cross-sectional area of an imaginary stream tube (which is constant in the free stream) *contracts* as the pressure drops with acceleration over the front portion of the wing (Fig. 4). The contraction of the stream tube due to acceleration is in opposition to the pressure drop which tends to expand a given volume of air.

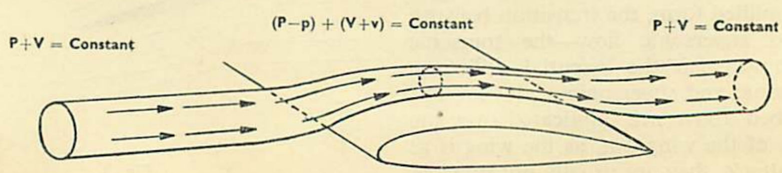

Fig. 4. Subsonic Concept of a Stream Tube

25. At subsonic speeds the contracting effect of the acceleration predominates. At supersonic speeds, however, the pressure drop becomes strong enough to reverse the tendencies, and once sonic speed is reached the stream tube begins to expand while it is accelerating. Fig. 5 shows in simple form the effect of increasing mach number on the cross-section of an imaginary stream tube.

26. An examination of Fig. 5 shows that for any given cross-section the mach number can be either subsonic or supersonic. It follows also that in the supersonic case, because the speed is higher, the pressure (and density) for a given cross-sectional area must be lower. In the transonic stage (while the shock wave is still on the wing) the flow changes from supersonic to subsonic through the shock wave, *i.e.* the conditions within the stream tube, at the shock wave, jump from those of point A in Fig. 5 to those of point B. The cross-section is the same at both points, but the speed at B is subsonic and the pressure (and density) considerably higher, although still below atmospheric.

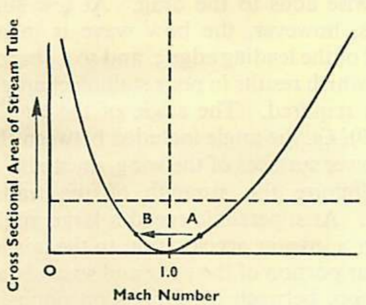

Fig. 5. Effect of Mach Number on the Cross-Sectional Area of a Stream Tube

Effects of the Differences between Subsonic and Supersonic Airflows

27. Examination of the table in para. 23 reveals that when a subsonic airflow passes over a corner, such as that shown in Fig. 6, the airflow decelerates suddenly with an accompanying increase in pressure ; at the same time the rising pressure acts on the boundary layer to slow it down, causing it to separate and break down the streamline flow. For this reason this type of sharp-edged corner is avoided on aircraft which are essentially subsonic.

28. When the airflow is supersonic it accelerates around the same corner and the pressure drops. Because the boundary layer is moving into a decreasing pressure gradient there is no separation, and streamline flow is maintained. Thus at supersonic speeds sharp-edged ridges can be used without any adverse effects on the airflow and hence on aircraft performance.

29. **Expansions.** Whenever the supersonic airflow is accelerated with an accompanying pressure drop, the effect takes place through what is termed a region of *expansion*. The region is bounded by two mach lines, whose slope is determined by the speeds and temperatures of the flows over each surface. *An expansion produces the opposite effect to a shock.* Whereas a shock wave decelerates the flow and increases the pressure, an expansion accelerates and decreases respectively.

30. Whenever the supersonic airflow is turned through an angle which decreases the speed and increases the pressure (contracting channel) the change is carried out through a shock, *with an accompanying increase in drag.*

31. Whenever the airflow is turned through an angle which increases the speed and drops the pressure (expanding channel) the change is carried out through an expansion, *with no drag penalty.* The first expansion is found at the most forward point of the sphere of influence of the corner (Fig. 6). The expansion then swings the flow around the corner, smoothly increasing the speed and decreasing the pressure until these values become proportional to the slope of the new path. The expansion then ceases and the accelerated flow continues parallel to the changed slope of the surface (Fig. 6).

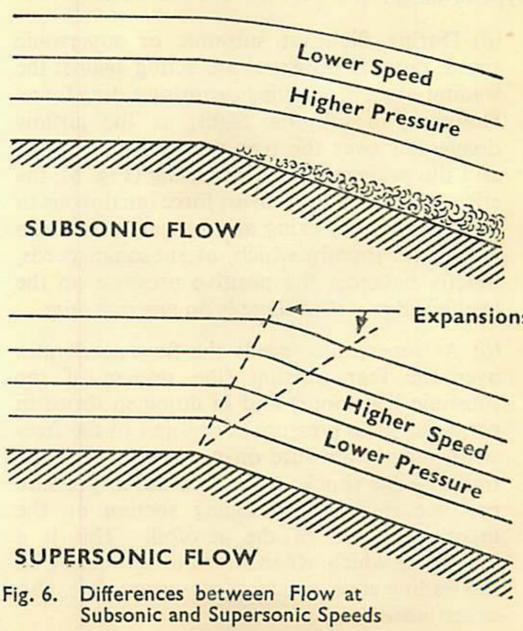

Fig. 6. Differences between Flow at Subsonic and Supersonic Speeds

32. **Aerofoil for Supersonic Flight.** For the reasons given in paras. 27 to 31 a wedge-shaped aerofoil section such as that in Fig. 7 can be used for supersonic flight. This shape of aerofoil, for a required thickness/chord ratio, gives least drag at these speeds. The bi-convex is another section that gives good results. However, the beneficial results of both sections at supersonic speeds are obtained at the expense of a poor subsonic performance, the lift that is obtained for a given wing area and air speed being low in comparison. All else being equal, this is reflected in higher landing and stalling speeds.

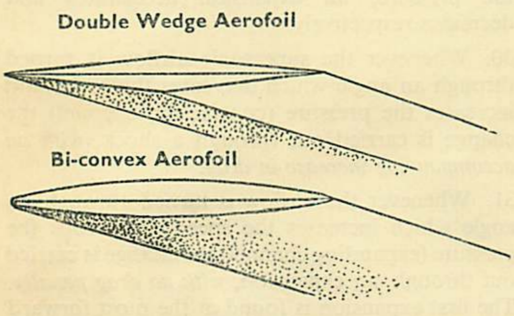

Fig. 7. Double-Wedge and Bi-Convex Aerofoils for Supersonic Speed

33. **Wave or Pressure Drag.** Wave drag at supersonic speeds is the major cause of the drag rise associated with flight at these speeds. Below is given a simple explanation of the origin of this type of drag :—

(*a*) During flight at subsonic or supersonic speed, positive pressures are acting against the leading edges of the wings, causing a drag force. However, in subsonic flight, as the airflow decelerates over the rear portion of the wing and the pressure starts increasing (Fig. 8), the effect is to produce a thrust force (analogous to the effect of squeezing an orange pip between finger and thumb) which, at subsonic speeds, exactly balances the positive pressure on the leading edge ; thus there is no *pressure drag*.

(*b*) At supersonic speeds the flow accelerates over the rear portion (the reverse of the subsonic behaviour) and in doing so drops in pressure. The pressure is restored to the free-stream static pressure on passing through the trailing-edge shock. The decrease in pressure produces a rearward facing suction on the tapering section of the aerofoil. This is a drag force which, when added to that acting on the leading edge, produces a *pressure drag*, also called *wave drag*.

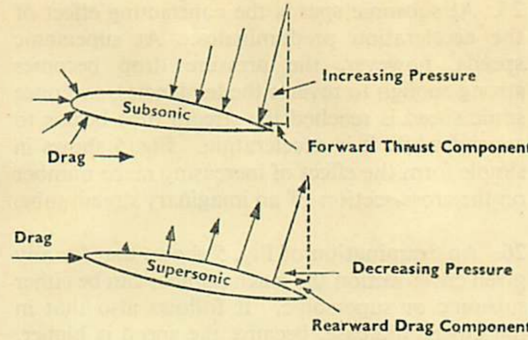

Fig. 8. Wave Drag

34. **Leading-Edge and Trailing-Edge Sharpness.** For supersonic cruising flight it is important from the point of view of drag that a sharp leading edge be used to reduce the strength of the pressure disturbance propagated forward from the leading edge. This makes for a weaker shock wave, giving less drag. The shock is practically attached to the leading edge, thus avoiding the subsonic region immediately in front of the nose which otherwise adds to the drag. At low supersonic speeds, however, the bow wave is in any case ahead of the leading edge ; and so a sharp leading edge, which results in poor stalling characteristics, is not required. The angle of the trailing edge (Fig. 9), *i.e.* the angle included between the upper and lower surfaces of the wing, must also be small to minimize the strength of the trailing-edge shock. At supersonic speed a large angle would impart a greater acceleration to the airflow over the rear portion of the wing and so cause a greater difference between conditions on opposing sides of the shock, *i.e.* a stronger trailing-edge shock forms to change the conditions over the wing back to free-stream conditions, and more drag results.

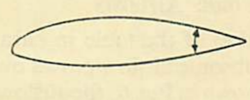

Fig. 9. Trailing-Edge Angle

35. **Camber.** Cambered aerofoils have adverse effects on drag and stability during transonic and supersonic flight. The consequent desirability of using symmetrical aerofoils means that the maximum lift coefficient at low speeds is decreased, leading to higher stalling speeds.

36. **Thickness/Chord Ratio.** The t/c ratio is a major consideration in the wing design of transonic and supersonic aircraft. The two most important effects of variation in the t/c ratio are dealt with below :—

(*a*) Thin aerofoil sections are the key to smooth transonic flight. Whereas the t/c ratio of the wings of transonic fighters is in the region of 8 to 10 per cent., that of supersonic aircraft is about 6 per cent. or less. The reason for these thinner wings can be seen in the curves of Fig. 10 which show the effect of speed on the drag coefficient of two wings, one of 12 per cent. and the other 7 per cent. t/c ratio. (It must be remembered that this coefficient is not the drag force itself but a term in the drag formula, $D = C_D \frac{1}{2} \rho V^2 S$. Variation in the C_D therefore affects the *rate* at which the drag increases with speed.) Note that the thinner wing has a much lower peak C_D—about a third of the thicker section. The curve shows that as the aircraft approaches mach 1·0, the drag coefficient, and so the rate of growth of total drag, increases rapidly. The C_D, which was constant at subsonic speeds, increases because of the wave drag and the adverse effects of the shock wave on the boundary layer. Above 1·0M the total drag continues to increase with speed, but because the C_D is falling the rate of increase is less. The drop in the value of the C_D immediately after 1·0M is mainly due to the decreasing transonic effects and improving boundary-layer conditions. At about 1·3M the C_D stabilizes at a higher figure than the subsonic value, the higher figure being caused by the effects of the wave drag characteristic of supersonic airflow.

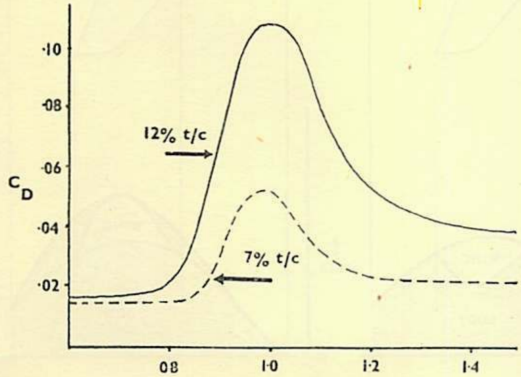

Fig. 10.
Variation of Drag Coefficient with Mach Number

(*b*) The t/c ratio plays an important part in the variation of C_L with mach number. When the critical mach number is reached the shock waves cause boundary-layer separation with its attendant loss of lift (decreased C_L). The C_L falls steadily to a minimum value and then rises to settle at a supersonic value slightly lower than the subsonic one corresponding to the particular angle of attack. The suddenness and degree of the loss in C_L depends largely on the t/c ratio. Fig. 11 shows typical C_L curves for a high and a low t/c ratio and indicates clearly the advantage of using a thin wing.

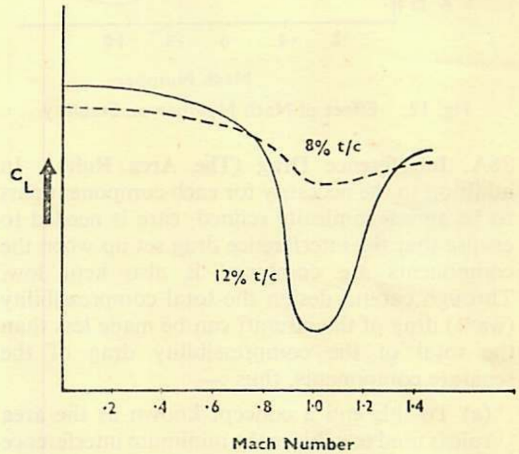

Fig. 11. Variation of C_L with Mach Number

(*c*) *Stability.* The lower part of Fig. 12 shows the effect of increasing speed on the fore-and-aft trimmed position of an aircraft using wings of 12 per cent. and 8 per cent. t/c ratio. The large nose-up pitch at about ·8M followed by the sudden change to nose-down is found on most aircraft using the comparatively thick aerofoils characteristic of aircraft that are basically subsonic. The same curve for the 8 per cent. wing shows a more gradual nose-down pitch followed by an equally gradual nose-up. These curves are illustrative only, since their shape varies with the aerofoil section used. The upper part of Fig. 12 shows the change in angle of attack necessary to maintain level flight (constant lift) with increasing mach number. Again, the superiority of the thin wing is obvious, the pronounced peak of the 12 per cent. section calling for an impossibly violent change of attitude.

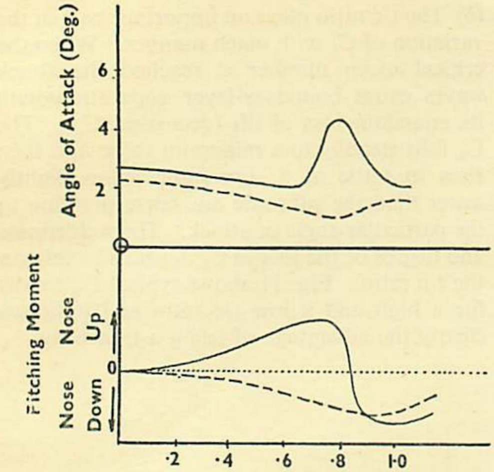

Fig. 12. Effect of Mach Number on Stability

36A. **Interference Drag (The Area Rule).** In addition to the necessity for each component part to be aerodynamically refined, care is needed to ensure that the interference drag set up when the components are combined is also kept low. Through careful design the total compressibility (wave) drag of the aircraft can be made less than the total of the compressibility drag of the separate components, thus :—

(*a*) To this end a concept known as the area rule is used to achieve the minimum interference drag. This rule offers advantages only over a fixed band of speed and its usefulness decreases on either side of this band. Broadly the area rule states that for minimum drag the variation of the aircraft's total cross-sectional area, along its length, should approximate to that of an ideally shaped object having minimum wave drag. This implies that the cross-sectional area profile should be fairly flat and free from sudden increases at the points where the other components are attached (Fig. 12A).

(*b*) In some aircraft the application of this rule is evident from the indented or " waisted " appearance of the fuselage where the fuselage cross-sectional area has been reduced so that the total cross-section does not result in marked departure from the required profile. On other aircraft having a high transonic drag caused through a poor cross-sectional area profile, an improvement may be obtained by carefully shaped fuselage " bulges ", fore and aft of the wing, which give a smoother profile.

(*c*) It should be noted that a waisted fuselage is not required if the aircraft is correctly proportioned from the outset. If a thin, swept-back wing is used, the rate of growth of the area profile can be kept close to the ideal without the structural complications of the waisted fuselage and the loss of internal space for fuel, etc.

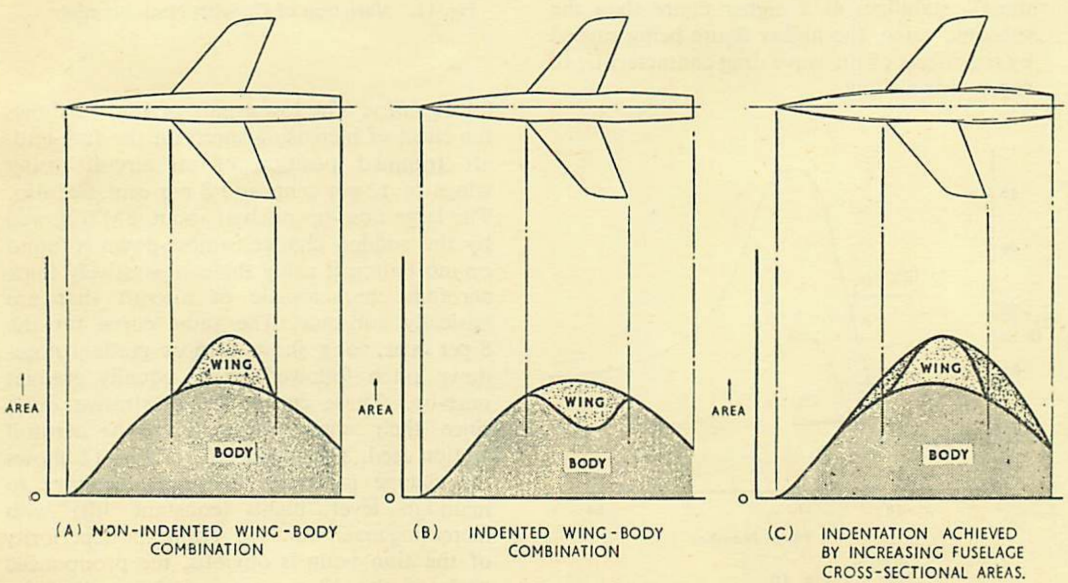

Fig. 12A. The Area Rule Concept

Effect of Angle of Attack on Supersonic Airflow

37. Fig. 13 shows, in simplified form, the streamline pattern of a supersonic flow around a double-wedge aerofoil at about mach 2, the section being set at zero (no lift) angle of attack. Initially the free-stream flow remains undisturbed until the air reaches the leading-edge shocks ; at this point the flow is turned parallel to the front surface of the wedge (contracting channel) with a resulting compression, an increase in pressure, and decrease in speed, as it passes through the shock. From Fig. 13 it can be seen that the streamlines remain parallel but are closer together over the front portion of the wing, indicative at supersonic speed of an increase in pressure (the reverse of the subsonic effect). The streamlines remain parallel until they reach the point of maximum thickness and the expansions above it. There the slope of the rear portion (expanding channel) causes a second change in direction, a consequent acceleration of the speed, and a drop in pressure. The streamlines are now more widely spaced, indicative of low pressure, and remain parallel to the upper portion of the rear of the aerofoil until the trailing-edge shock is encountered. At this point the shock wave turns the flow so that it joins the free stream without discontinuity ; at the same time it is slowed down and the pressure restored to the free-stream value. Since the aerofoil is symmetrical and at zero lift, the same pattern is repeated below the wing.

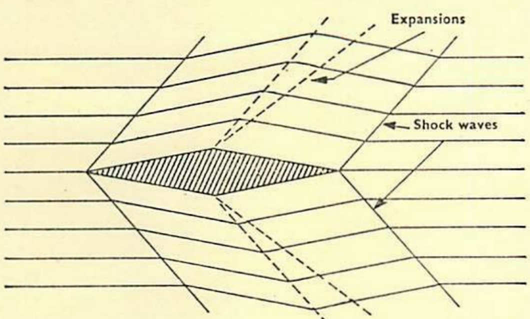

Fig. 13. Supersonic Flow Around a Double-Wedge Aerofoil at Zero Angle of Attack

38. Fig. 14 shows the same aerofoil at a high angle of attack. Here the whole of the upper surface is inclined away from the free stream and is thus having the effect of an expanding channel while the whole lower surface is inclined towards the free stream and is having the effect of a contracting channel. When the supersonic free stream reaches the sphere of influence of the wing, the airflow over the upper surface passes through expansions at the leading edge and is accelerated with an attendant pressure drop. The amount of acceleration and drop in pressure is proportional to the slope of the upper surface. The streamlines again remain parallel until they encounter the effects of the point of maximum thickness. Here the expansions swing the streamlines around parallel to the increased slope of the rear surface with a further increase in speed and drop in pressure. At the trailing edge, conditions are instantaneously restored to those of the free stream through the thickness of the shock wave. There is, therefore, a region of low pressure over the entire upper surface.

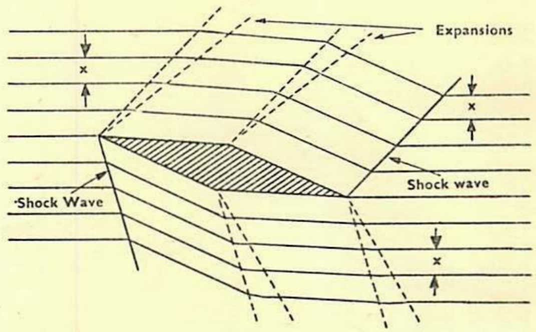

Fig. 14. Supersonic Flow Around a Double-Wedge Aerofoil at Positive Angle of Attack

39. The lower surface being inclined to the free stream, the airflow is *decelerated* through the shock from the leading edge, and the pressure rises. Conditions remain constant until the point

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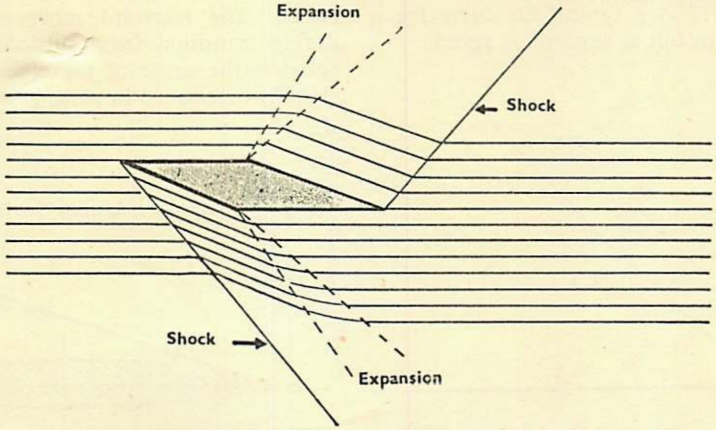

Fig. 15. Supersonic Flow Around a Double-Wedge Aerofoil at Angle of Attack for Best L/D Ratio

of maximum thickness is reached. At this point, the surface of the aerofoil slopes away from the streamlines over the front, giving the effect of a less rapidly contracting channel. Therefore an expansion occurs with an increase in speed and a drop in pressure ; but the speed and pressure are still lower and greater respectively than those of the free stream, and so a positive pressure is still being experienced over this portion of the wing. The expansion at the trailing edge restores the speed and pressure there to the free-stream values.

40. Notice that both expansions and shocks serve to bend the streamlines. Therefore by giving the wing a positive angle of attack, the air is deflected downwards to obtain lift, but by a very different means from that employed in the case of the subsonic wing, the downwash being confined to the region between the respective shocks and expansions.

41. **Best L/D Ratio.** Fig. 15 shows a double-wedge aerofoil at the angle of attack for best L/D ratio. This angle is equal to half the angle of the wedge and it can be seen that the upper front and lower rear surfaces are parallel to the airflow. Consequently the free air stream is virtually unaffected by these surfaces and is only acted upon by the two inclined surfaces. Therefore around the corner of the upper surface the flow is accelerated through the expansions and swung round parallel to the rear portion with a drop in pressure ; at the trailing edge the shock restores the pressure and velocity to free-stream conditions. On the lower surface at the leading edge another shock indicates the pressure rise found on the lower front portion, and an expansion at the corner restores the higher pressure and reduced speed to free-stream values. Thus only two shock waves are attached to the wing at this angle of attack and the wave drag is therefore less than that at any other angle of attack, at which all four planes of the section are affecting the flow.

42. The mechanics of supersonic lift are simpler than those of subsonic because the pressure at any point over the wing is dependent only on the slope of the surface over which it is passing. If the slope is towards the free stream the pressure is always positive, if away it is always negative. Pressure can therefore be changed only by varying the angle of attack.

43. The theoretical pressure distribution of a double-wedge aerofoil at a high angle of attack, divorced from boundary-layer and other effects, can therefore be shown as two sets of parallel lines (Fig. 16), the area between the lines of the upper and lower surfaces being proportional to the lift. The C.P. will be at the 50 per cent. chord

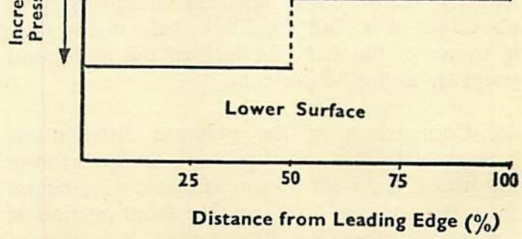

Fig. 16. Pressure Distribution of a Double-Wedge Aerofoil at Moderate Angle of Attack at Supersonic Speed

position. Fig. 17 is a typical lift curve for a wedge-shaped aerofoil at supersonic speed.

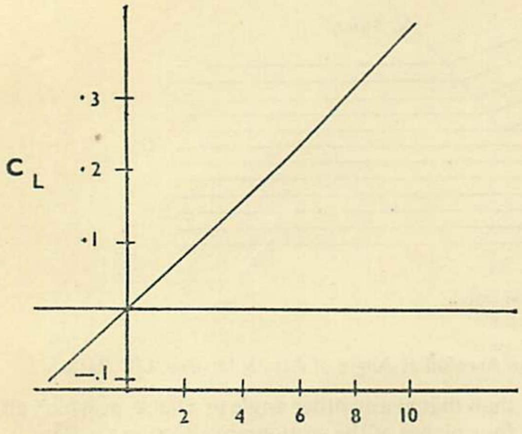

Fig. 17. Lift Curve of Typical Double-Wedge Aerofoil at Supersonic Speed

Bi-Convex Aerofoil

44. The foregoing paragraphs have dealt with flow about a double-wedge section because of the ple nature of the flow about it. The bi-convex section, although subject to the same effects, has, by reason of its curved section, a different pressure distribution.

45. The outlines of the bi-convex section are arcs of circles, therefore the point of maximum thickness is at the 50 per cent. chord position. Because the slope of the section is changing continuously the pressure must follow suit and the separate expansions of the double-wedge section now occur progressively along the surface. Fig. 18 shows the theoretical pressure distribution of this section at a fairly high angle of attack. The dotted curve for the top surface is a falling-off in lift that occurs in practice through adverse boundary-layer effects, while the dotted line for the lower surface shows a local increase in pressure near the leading edge. Thus the total lift remains about the same, but the effect of the dotted lines is to move the C.P. forward of the mid-chord point by about 10 per cent.

46. Comparison of the pressure distributions between subsonic and supersonic airflows shows that most of the lift on the subsonic wing comes from the pressure drop over the front portion of the upper surface (Fig. 19), *i.e.* the C.P. is situated at about the quarter-chord point. The C.P. of the supersonic wing is almost at the half-chord point. The rearward movement of the C.P. during transition from subsonic to supersonic speed is the cause of the changes in trim and stability mentioned in para. 36.

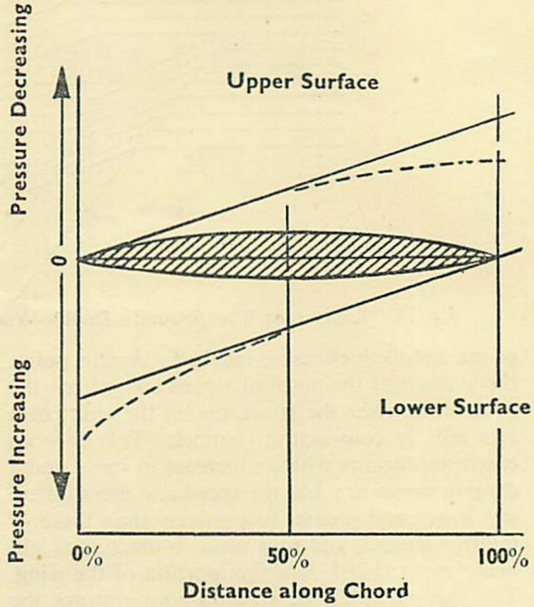

Fig. 18. Pressure Distribution of a Bi-Convex Aerofoil at Moderate Angle of Attack at Supersonic Speed

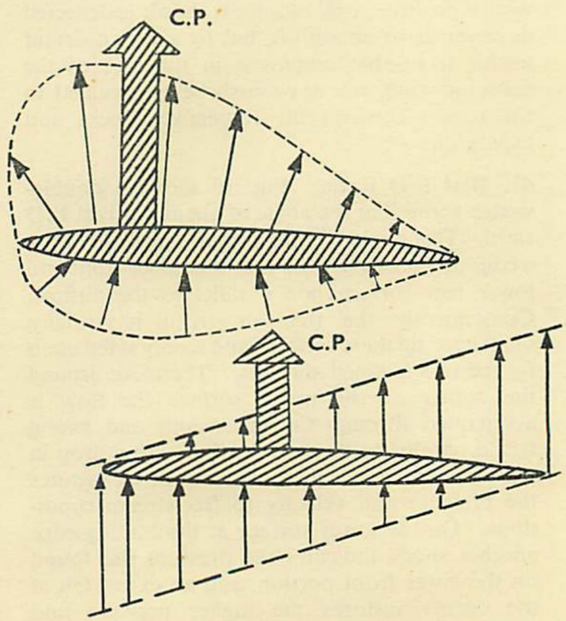

Fig. 19. Comparison of Subsonic and Supersonic Pressure Distribution

Sweepback

47. By the use of sweepback, wings having a basically subsonic aerofoil section can be used in the transonic and supersonic speed ranges. Whereas the unswept wing of medium t/c ratio (about 8 per cent.) is subject to the limitations imposed on it by the formation of shock waves and the attendant shock stall, the same wing when swept back at an angle to the airflow shows a marked improvement. The following paragraphs explain the effect of sweepback.

48. If a length of wing (Fig. 20) is placed in a wind tunnel running at a given speed, say just below the critical mach number, then a certain airflow pattern is set up about the wing. Any increase in the tunnel speed would therefore affect the pattern of the streamlines, in particular any increase would form the first shock waves on the wing.

49. If the wing were now pulled spanwise through the tunnel, the chordwise pressure distribution would be unaffected even though the resultant velocity from the tunnel and spanwise motion components was greater than the tunnel speed, *i.e.* only the chordwise component determines the flow pattern over the aerofoil. If the wing were pulled spanwise at the tunnel speed the resultant speed (from Pythagoras) is $\sqrt{2}$, or 1·4 times the individual speeds in a direction at 45° to the axis of the tunnel. If the same wing were mounted with 45° of sweep in an airflow the aerodynamic result would be the same. If an aircraft with 45° of sweep were flying at 600 kts., the wing would behave in the same way as an unswept wing at about $600/1{\cdot}4 = 430$ kts. The effect of sweepback can therefore be summarized by stating that it reduces a high subsonic or even supersonic forward velocity to one spanwise component which does not affect the pressure distribution, and a second component acting at 90° to the leading edge which produces a subsonic pressure distribution having none of the pressure drag which accompanies shock waves. This holds good only so long as the 90° component is less than the critical mach number.

50. **Effect of Sweepback on C_L.** Since at any given speed the pressure distribution of a swept wing is equivalent to that of an unswept wing working at a lower speed (the difference depending on the amount of sweepback) the lift coefficient and therefore the total lift of a swept wing is less than that which would be developed by an unswept wing at the same speed. The greater the angle of sweep the greater is the loss of lift when compared to a wing with no sweep. The reduced C_L is a natural effect when it is remembered that only a *component* of the free-stream velocity, and not the full velocity, is determining the pressure distribution.

51. Since the wing is swept back, the effective chord is greater than the true chord measured at 90° to the leading edge. The t/c ratio is thereby improved and additional benefit derived from this consideration. It is, however, basically the effect explained in para. 48 that is responsible for the improved characteristics.

Effect of Compressibility on Trailing-Edge Control Surfaces

52. Any movable control surface which is hinged behind a fixed surface suffers a marked reduction in effectiveness at high mach number. Ideally, when the control surface is moved it should influence the pressure distribution over both itself and the surface ahead, but during the transonic stage its influence is restricted to the area aft of the shock wave which springs from the fixed surface ahead of it; consequently the effectiveness of the control is reduced and larger movements are necessary to obtain a given reaction from the aircraft.

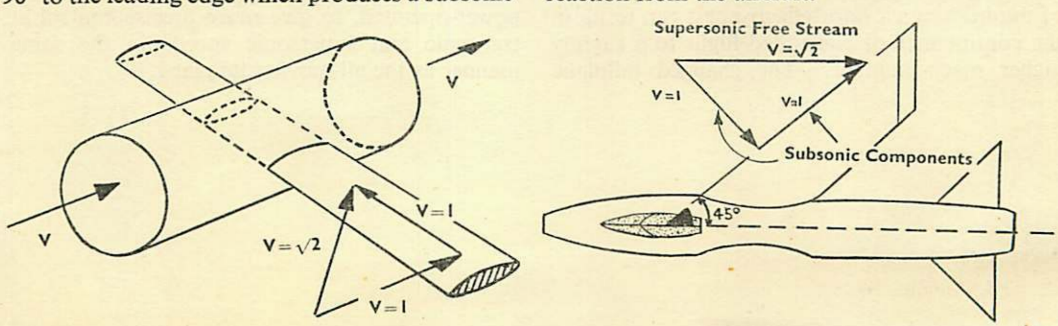

Fig. 20. Effect of Sweepback

53. A further consideration, which also decreases effectiveness, is the effect of the shock on the boundary layer. As explained earlier, the boundary layer usually separates immediately after the shock and flows straight back instead of following the contour of the surface. The outcome is that the control surface is operating in a region of air that is aerodynamically dead, being stalled and turbulent. Again, this means that larger control movements are required to produce a given reaction from the aircraft.

54. In practice the sum effect of paras. 52 and 53 is found to be a reduction in the elevator effectiveness, starting at the critical mach number and becoming worse as the mach number increases. In some instances a total loss of elevator control may occur which is only regained after the aircraft has descended to an altitude where the temperature is higher and the mach number lower. Decreased elevator effectiveness is apparent (in the absence of powered controls) by the controls becoming heavier to operate. At the same time, irrespective of the type of control system, the response of the aircraft becomes sluggish, large control column movements being necessary to change the attitude.

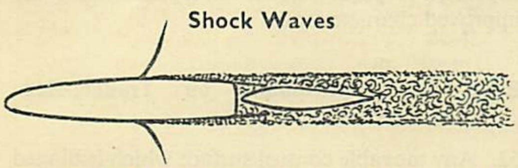

Fig. 21. Effect of Shock Stall on Control Surface Effectiveness

Variable-Incidence Tailplanes

55. A limited amount of control can be obtained by the use of variable-incidence (V.I.) tailplanes. These are primarily intended as a means of trimming the aircraft, but their use in the presence of diminishing elevator effectiveness can result in the continuance of controlled flight to a slightly higher mach number. The changed tailplane incidence, unaffected by any fixed surface ahead of it, causes a change in the complete pressure distribution over the surface and is thus much more effective over the transonic period than a hinged control surface. The disadvantage of the V.I. tailplane for normal flight control is that movement of the control column must be supplemented by the operation of a separate tailplane incidence-changing switch.

All-Moving or Flying Tails

56. To overcome the deficiencies outlined in paras. 52 to 55 the flying or all-moving tail is adopted. With this system the tailplane is made the primary control surface and is therefore coupled directly to the control column. A downward movement of the leading edge reduces the angle of attack and tailplane lift and causes a nose-up force on the aircraft. In this way full control can be maintained throughout the transonic period, the tailplane providing a sensitive and powerful control at all speeds.

57. With a flying tail the elevator can be dispensed with entirely leaving a single surface of the required area. On some aircraft, however, the elevator is retained and linked to the tailplane in such a way that it moves in the appropriate direction to assist the action of the tailplane ; for example, a forward movement of the control column increases the angle of attack of the tailplane and, by virtue of the linkage, moves the elevator downwards.

All-Moving Wing Tips

58. In the same way as the all-moving tail improves longitudinal control, so all-moving wing tips overcome the aerodynamic weakness of ailerons which operate behind a fixed surface. The extreme tip of the wing is hinged, and usually power-operated, to give more precise control at transonic and supersonic speeds in the same manner as the all-moving tailplane.


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