A STUDY OF SOME PROBLEMS AFFECTING THE DESIGN OF WINGS FOR TRANSONIC AIRCRAFT

Thesis by

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Part I. Summary

A survey of the literature concerning the design of wings for transonic aircraft is made. The problems investigated were 1) for the transonic wing: reduction of drag at high subsonic Mach numbers; achievement of high maximum lift coefficient and stability at low speeds; and determination of air loads in compressible subsonic flow, 2) for the supersonic wing: determination of flow fields at transonic Mach numbers; achievement of adequate low speed performance of a supersonic wing; and determination of air loads at transonic speeds, and 3) structural problems of both types of wings. The extent of present knowledge is outlined, and future fields of desirable research are indicated. A bibliography of reports pertaining to transonic wing design, accompanied by brief summaries is given. A supplementary bibliography of other reports of general interest to transonic wings is also presented.

Part II. Introduction and Statement of the Problem

Prior to World War II, when the only practical method of propulsion of aircraft was the screw propeller, there was little expectation of flight at speeds higher than the speed of sound in the foreseeable future. With the recent development of jet and rocket engines for aircraft, however, the possibility of flight at transonic and supersonic speeds has become an immediate reality. Because supersonic aerodynamics is amenable to mathematical investigation, theoretical work has been done in this field for some time, and basic laws are well developed. In the transonic range, however, airflow exhibits indeterminate characteristics, and not very much real knowledge is available, either theoretical or experimental. It has been said that the state of transonic aerodynamics today is roughly comparable to that of subsonic aerodynamics before Prandtl developed his famous three-dimensional wing theory. Since it is obvious that an airplane must fly in the transonic zone if substantial speed increases over present day practice are to be achieved, much work remains to be done before safe and practical advances are made.

The purpose of this study is to further future research in the transonic field by collecting and evaluating as much as possible of present knowledge concerning the design of perhaps the most important and critical member of the transonic airplane, the lifting wing. For convenience in this study, transonic aircraft have been divided into two classes: 1) the transonic airplane, in which the wing flies at subsonic Mach numbers, while the fuselage flies at supersonic Mach numbers; and 2) the supersonic airplane, in which the entire airplane is assumed to fly at supersonic Mach numbers.

Part III. Historical Background of Transonic Research

A large part of the research reviewed in this survey was contributed by German laboratories immediately before and during the recent war. Individual reports are reviewed in the bibliography, and the laboratories and their most prominent men will be mentioned only briefly:

1. Aerodynamische Versuchsanstalt Göttingen. Also referred to as AVA. Principal contributors were Hansen, Lemme, Schwenk, Brennecke, Krüger, and Prandtl.

2. Technische Hochschule Braunschweig. Principal contributor was Jacobs.

3. Technische Hochschule Hannover. Principal contributor was Luetgebrune.

4. Deutsche Versuchsanstalt für Luftfahrt, Berlin-Aldershof. Also referred to as DVL. Principal contributors were Thiel, Weissinger, Puffert, and Göthert.

5. Technische Hochschule Karlsruhe. Principal contributor was Sauer.

6. Volkenrode. Principal contributors were Busemann and Guderley.

Other research work in high speed aerodynamics was carried out during the war in England, especially in the Royal Aircraft Establishment at Farnborough and the National Physical Laboratory at Teddington, and in the United States at the NACA laboratories and several universities, but these tests were largely on specific production type models with compressibility troubles, and did not constitute a well integrated attack on the problems of transonic flow as did the German tests.

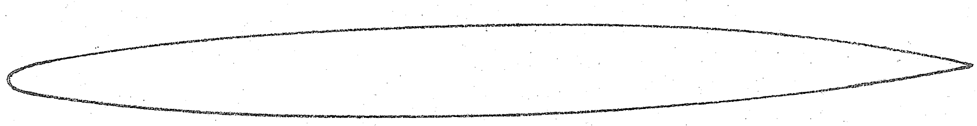
As Germany was occupied near the end of the European war, the capture of German documents caused a great awakening in American and English research circles. As many documents as could be found were rushed back to America for translation and study, for the United States realized that it was far behind in the transonic field. German tests were repeated in NACA laboratories (see recent NACA Technical Notes in the bibliography), and future research has been planned. This work is being done under the combined auspices of the NACA, the Army and Navy air forces, and the various aircraft companies such as Bell and Douglas. Basic research is being carried out by the NACA tunnels and the various universities, while test models such as the Bell XS-1 and XS-2 and the Douglas D-558 Skystreak are being built to obtain experimental flight test data. Many other projects unknown to the author are undoubtedly being carried out by other companies under military security. The program is still only in its infancy, however, and none of the projected transonic airplanes has as yet attempted to exceed Mach number 1. The British are also carrying out a high speed research program, but nothing could be learned about it except the announcements of high-subsonic jet fighters which have appeared from time to time in the press.

Part IV. Aerodynamical Problems of the Transonic Aircraft Wing

Reduction of Drag at High Subsonic Mach Numbers

The primary obstacle to flight at speeds higher than 500 miles per hour is the sharp increase in drag as the airplane Mach number approaches the speed of sound (Ref. 44). This abrupt increase in drag is usually close to the Mach number at which the local Mach number on some part of the wing surface reaches 1, but is also associated with other high speed phenomena such as boundary layer separation and excessive turbulence (Ref. 29, 30). The point of the sharp upturn of drag is usually referred to as the critical Mach number of the airplane.

There are several methods used to suppress this drag increase at high speeds. One of the first was the use of the so-called "low-drag" airfoils such as the NACA 16 series (Ref. 22). These airfoils are characterized by a small radius of curvature at the nose, increasing in thickness to a maximum thickness place much farther back toward the trailing edge than in other subsonic airfoils.



NACA 16-009 Airfoil

The relatively long chordwise section of slowly increasing thickness thus maintains the favorable pressure gradient conducive to laminar boundary layer over as much of the wing chord as possible, suppressing the compressibility burble, and raising the critical Mach number. Another method of keeping the boundary layer laminar is the use of extensive boundary layer removal on the wing surface, using the jet engines of the airplane as a pump (Ref. 20). The Germans have investigated this feature, and say that it looks promising except for the resulting mechanical complication. It should also be mentioned here that all high speed airfoils are constructed with as smooth a surface as possible to help maintain a laminar boundary layer (Ref. 31, 32).

Still another design feature which increases the critical Mach number is the use of very low aspect ratio wings (Ref. 33). Göthert has investigated this idea, and believes that the observed drag reduction at high subsonic Mach numbers of low aspect ratio wings is due to tip effects extending along the entire wing span. In view of the fact that low aspect ratio wings are very favorable from a structural standpoint because of the high wing loadings encountered in high speed flight, transonic wings will probably be of lower aspect ratio than is common in subsonic design.

By far the most promising method of increasing the critical Mach number of a transonic wing is the use of sweep-back or sweep-forward (Ref. 1 et seq). The use of sweep to reduce drag at high speeds was intensively investigated by the Germans during the recent war without the knowledge of American or English engineers. The method by which drag is reduced by sweep can be roughly explained by saying that the effective Mach number of a swept wing is a function only of the component of flow normal to the leading edge of the wing, or:

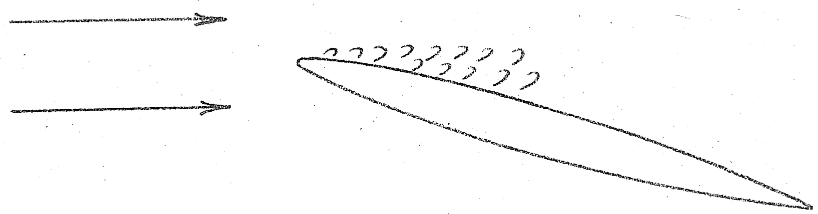
$$M_{WING} = M_{AIRCRAFT} \cdot \cos \varphi$$

Where φ is the angle of sweep. If the angle of sweep is 45°, it is seen that the critical Mach number is raised by over 40% in the first approximation, which is a very worthwhile increase. It is by the use of sweep that transonic airplanes can be designed to fly at supersonic speeds while the wings are still experiencing subsonic flow.

The use of sweep has several rather serious drawbacks, however. One of the most serious is the rather unsatisfactory condition of the boundary layer on the wing towards the trailing edge and the tips (Ref. 2). This excessive boundary layer growth is caused by the spanwise flow over the wing resulting from the tangential component of air flow which we neglected in the previous effective Mach number formula. The effect of this thick boundary layer at high speeds is two-fold. First, it greatly impairs the effectiveness of control surfaces such as ailerons or spoilers which are placed on the trailing edge at the wing tips (Ref. 7). This effect is so pronounced that unless slots or some other method of preserving air flow at the tips is used, ailerons cannot be assumed to give adequate control at high speeds. The second effect is the lack of adequate longitudinal and lateral stability at high subsonic speeds. As will be seen in a later section on air loads, a large part of the load on a swept-back wing is carried on the tip sections. If the air flow over these sections is subject to violent fluctuations as separation and shock stall take place, excessive pitching moments will be exerted on the airplane because of the fact that the tips of a swept wing have a large moment arm about the airplane center of gravity in both longitudinal and lateral directions. These difficulties have not as yet been completely overcome. The swept wing also shows undesirable stability characteristics at low speeds, which will be discussed in the following section.

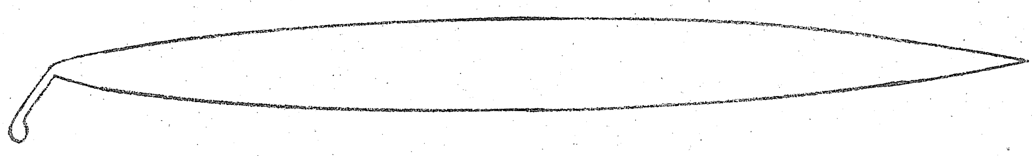
Achievement of High Maximum Lift Coefficient and Stability of a Transonic Aircraft Wing at Low Speeds.

A problem common to all high speed wings is the achievement of a satisfactory maximum lift coefficient at low speeds for landing and take-off. As stated in the preceding section, a primary requirement for a high speed wing is a small radius of curvature of the leading edge. When such a wing is held at a large angle of attack, the sharp leading edge causes a very steep pressure gradient above the nose which leads to early separation on the upper surface with resulting loss of lift.



NACA 16-009 Airfoil at High Angle of Attack

This separation can be reduced somewhat by the use of conventional slots or slats, which serve to turn the air around the sharp nose and force it along the top surface of the airfoil (Ref. 10). A much better method, however, is the use of a special nose flap developed by the Germans during the recent war (Ref. 11, 12). This nose flap appears:



NACA 16-009 Airfoil with Nose Flap

As can be seen, the nose flap tends to round the contour of the upper surface at the nose and to flatten out the severe adverse pressure gradient, preventing separation. Experimental results show that it is by far the most effective maximum lift coefficient increasing device of any available thus far, and it is doubly effective in that it works very well with the more common split flap, which alone loses a great deal of its effectiveness because of the thick boundary layer present on the trailing edge of a swept wing. The nose flap, unlike more conventional flaps, has an optimum angle of deflection for maximum lift coefficient, larger or smaller deflections resulting in undesirable air flow over the wing. For this reason, the flap actuating mechanism must be quick acting in order that the flap can reach optimum deflection before the intermediate loss of lift can seriously affect the flight path of the airplane. The design of a suitable nose flap together with its actuating mechanism presents a fairly difficult mechanical problem, but in view of the extremely beneficial aerodynamic effects of the flap, it would seem that serious efforts to develop the nose flap are warranted.

In addition to the difficulty of obtaining a high maximum lift coefficient, the swept wing also suffers from some undesirable stability characteristics at low speed. As stated before, separation and stalling occurs at the region of highest loading, which is the tip section in the case of the swept-back wing. As is the case at high speeds, this separation gives rise to large pitching moment changes, which are especially dangerous at low speeds since the airplane will probably be taking off or landing, and so will be close to the ground.

Besides the stalling difficulties, pronounced sweep changes the effective dihedral of a wing (Ref. 9), and so alters the stability characteristics in unstalled flight. The swept-back wing exhibits an effective dihedral large enough to be objectionable, while the swept-forward wing is somewhat unstable. For this reason, together with other design considerations, the swept-back wing is used more prevalently than the swept-forward. The very large effective dihedral caused by sweep-back may be reduced somewhat by built-in negative dihedral, but only at the expense of a reduced maximum lift coefficient and lift curve slope. Lemme (Ref. 6) experimented with various truncated swept wings in an attempt to obtain the benefits of sweep without the undesirable stability characteristics and claims some success, but little application seems to have been made elsewhere of his work.

Determination of Air Loads in Compressible, Subsonic Flow

In the structural design of an aircraft wing, the first thing a designer must know are the air loads to be expected in all conditions of flight. Low speed air loads can, of course, be computed by incompressible flow methods already in standard use. For the case of a straight wing at high subsonic Mach numbers, the Prandtl-Glauert or Karman-Tsien methods as explained by Liepmann (Ref. 44) and extended by Göthert (Ref. 72) are useful in conjunction with the many standard methods of calculating incompressible lift distribution. The Prandtl-Glauert and Karman-Tsien methods consist essentially of multiplying the incompressible pressure coefficient, C_{p_o}, by a correction factor:

$$C_{P_M} = \frac{1}{\sqrt{1-M^2}} C_{P_0}$$

for the Prandtl-Glauert method, and:

$$C_{P_M} = \frac{C_{P_0}}{\sqrt{1-M^2} + \frac{M^2}{1+\sqrt{1-M^2}} \cdot \frac{C_{P_0}}{2}}$$

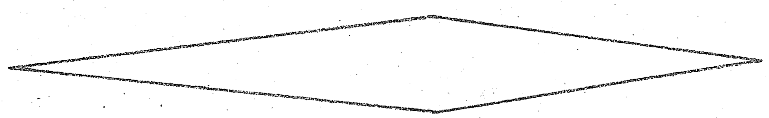
for the Karman Tsien method.

For the swept wing, however, the standard straight wing methods give inaccurate results. Doris-Cohen (Ref. 15) has developed a method of calculating the pressure distribution over a swept elliptical wing, but this method is not very flexible. Two theories have been developed in Germany, one by Multhopp, and a somewhat better one by Weissinger (Ref. 17). These methods have been subjected to experimental check (Ref. 18), and Weissinger's method gives good agreement with observed results. They are essentially extensions of the Prandtl three-dimensional wing theory. All three of these theories show that lift distribution moves towards the tips on a swept-back wing, and towards the root on a swept-forward wing. It is at these points of high pressure that separation and stalling first occur, as mentioned in the previous section. It should be emphasized that these methods apply only to flow in which the Mach number is subsonic.

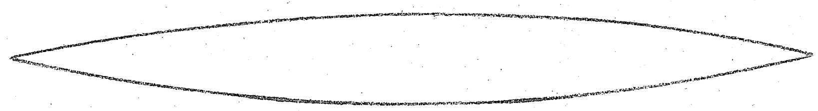
Part V. Aerodynamical Problems of the Supersonic Aircraft Wing

Nature of the Supersonic Wing Shape

In the case of an airplane designed to fly at supersonic speeds, the main design considerations will, of course, be dictated by the theory of supersonic flow, altered only to the extent of insuring that the airplane will be able to pass through the transonic zone with reasonable ease and safety. Supersonic airfoil theory is beyond the scope of this paper, but the nature of the requirements for a satisfactory supersonic airfoil can be briefly summarized (Ref. 44). In order to keep drag to a practical value, the leading edge of the airfoil must be a sharp wedge with as low an included angle as possible. The trailing edge is, of course, also a sharp wedge, but the shape of the airfoil in between these sharp edges is not so critical as in subsonic airfoils. Two types in favor are the diamond shaped airfoil



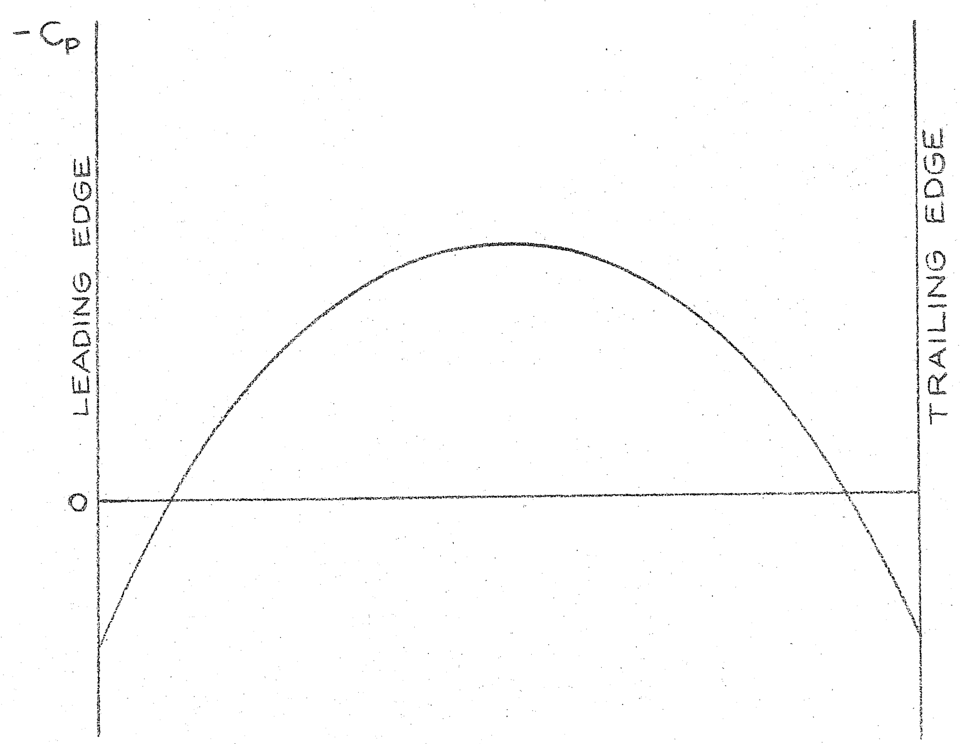
and the lenticular, or double convex airfoil.



The airfoil will be as thin as possible, because the wing with the highest lift drag ratio in supersonic flow is the one with zero thickness. At supersonic speeds, flow is generally steady, with oblique shock waves and expansion zones trailing back from the sharp leading and trailing edges in order to turn the flow along the airfoil. The wing will not necessarily be swept, as authors seem to disagree on the merits of this type of plan form at supersonic speeds. Aspect ratio will probably be fairly low for a number of reasons, such as difficulty of passing through the transonic range, and structural efficiency.

Determination of Flow Fields at Transonic Mach Numbers

This study is concerned with the performance of such a wing as it moves across the transonic range. It is in this region of turbulence, non-stationary shock waves, and buffeting that the greatest unsolved problems of high speed aerodynamics lie. As stated in Part IV, present theory enables one to calculate the pressure distribution over a wing as long as the local Mach number is below 1. In this region, the pressure distribution of a typical symmetrical wing is continuous and appears:

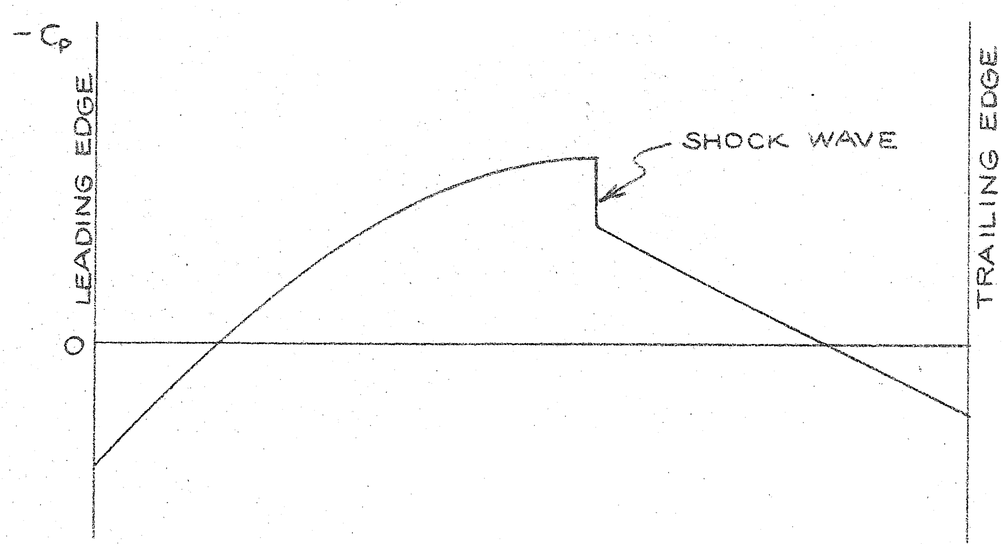


At Mach number 1, however, previous theory based on the Prandtl-Glauert analysis breaks down, as it is based on the assumption that the local Mach number is everywhere below 1. A small perturbation theory (Ref. 44) which assumes that the Mach number everywhere is close to 1, but either above or below it, has been developed, but this analysis is of questionable value. It gives rise to potential equations of the type.

$$\frac{\partial \phi}{\partial x} \cdot \frac{\partial^2 \phi}{\partial x^2} - K \frac{\partial^2 \phi}{\partial y^2} = 0$$

It is seen by inspection that this equation is non-linear, and of a type for which no general mathematical solutions are now known. The equation must thus be solved numerically at specific points -- at best a laborious process, and in this case, a dangerous one. It is known that transonic flow fields may exhibit discontinuities such as shock waves in the physical field and limiting lines in the mathematical analysis. Single numerical points in such a field may thus lead to distorted and inaccurate interpretations.

At some point, depending on flow conditions, potential flow is broken down by the formation of a shock wave, and the pressure distribution on an airfoil becomes



In this region there exist not even a questionable theory by which flow conditions may be calculated. Experimental measurements may be used to obtain data, but a serious drawback to this approach aside from practical difficulty is that there exists no proof that a particular flow pattern in a transonic region is unique (Ref. 44). In fact, experiment has consistently shown that the shock waves in this region are extremely nonstationary (Ref. 40). Some observers feel that these variations are intrinsic in the nature of the flow, but others think they are the result of minor variations in physical parameters which are critical to the flow pattern, such as Mach number and angle of attack. Liepmann (Ref. 44) believes that angle of attack is very critical and that the non-stationary flow is largely a result of torsional deformation of the model wing, or of change of direction of flow in the tunnel. If this is the case, then aerodynamic forces on an airplane wing may depend on the structural rigidity of that wing, and a situation akin to flutter may be set up. The flutter problem is further complicated by the fact that some boundary layer eddies at transonic speeds are so large that they may have characteric frequencies of the order of the vibrational frequencies of the wing itself. Boundary layer flow is in turn highly dependent on surface imperfections of the wing (Ref. 31, 32) which may again be dependent on the deformation of the wing under load. It can be seen that the buffeting problem at transonic speeds is not easily solved.

Looking at the effect on the wing as a whole, it has been found that the shock wave generally forms on the upper surface first, causing an abrupt loss of lift and change of pitching moment resulting in longitudinal stability difficulties. At a slightly higher Mach number, a shock will form on the lower surface, causing an increase in lift and another change in pitching moment, again disturbing the equilibrium of the airplane. In addition, when it is realized that these shock waves are non-stationary, it is seen that the maintenance of adequate control while passing through the transonic range is indeed a very serious problem which has not as yet been solved. Present hopes are that buffeting can be greatly reduced by accelerating rapidly through the transonic range, as in a steep dive under full power. Even this method is open to some doubt, however, as large accelerations may materially increase the aerodynamic forces at Mach number 1. Such a dive has not been attempted at the time of writing, and it is therefore not known whether the method will work satisfactorily.

Achievement of Adequate Low Speed Performance of a Supersonic Wing

The low speed performance of an unswept supersonic wing does not present stability problems so great as those of the subsonic swept wing, but the problem of obtaining a satisfactory maximum lift coefficient is still present. The sharp leading edge of the supersonic airfoil causes severe separation at high angles of attack, and it appears that a device such as the nose flap (Ref. 12) to change the airfoil contour is a necessity. The nose flap presents serious structural difficulties, however, which will be discussed in detail in Part VI.

Determination of Air Loads at Transonic Speeds

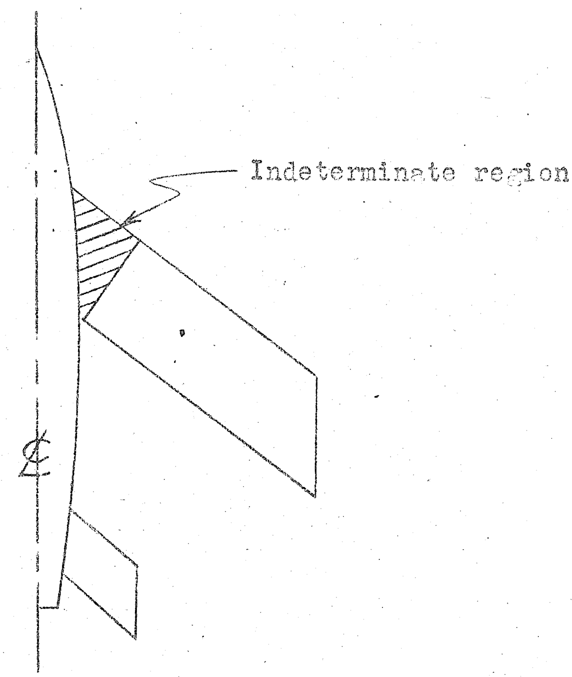
Owing to the absence of a satisfactory theory in the transonic range, the only method at all practical at the present time of obtaining air loads is by experimental measurement, and even this method is very unsatisfactory because of the unsteady nature of the flow field. Present practice seems to consist of using large factors of ignorance and hoping for the best. In line with this policy, the Bell XS-1 and Douglas D-558 are stressed to withstand an acceleration of 18g (Ref. 135-137). It is obvious that a large amount of work remains to be done in this field before an efficient design based on reliable calculations can be made.

Part VI. Structural Problems of the High Speed Wing

Summarizing the aerodynamic requirements of Parts IV and V, the high subsonic Mach number wing will have a thin, low-drag airfoil with a relatively sharp leading edge and maximum thickness at about mid-chord, and will probably have a large angle of sweep. The supersonic wing will have a thin airfoil with pointed leading and trailing edges, and perhaps a diamond-shaped cross section. The wing will not necessarily have sweep. In both cases, the wing must exhibit a very smooth surface under all operating loads to prevent disturbance of the boundary layer, and in each case the wing must have auxiliary lift-increasing devices such as flaps on the leading and trailing edges, or slots, or boundary layer removal. The wing must have a very high load factor, at least until a more satisfactory method of determining air loads has been obtained, and must also have relatively high rigidity in both bending and torsion to avoid flutter (Ref. 45, 46). These are indeed a most difficult set of specifications to meet.

Since the wing must be very thin, and at the same time have a very smooth surface, high strength, and great rigidity, it seems obvious that wing design must abandon the conventional stiffened skin construction of the past ten or fifteen years and go to a full monocoque design with rather thick skin, amounting, in fact, to a shell made of plates which will never be allowed to go into the buckled regime. The stress analysis of such a structure can be made by standard plate analysis techniques except for a few special problems.

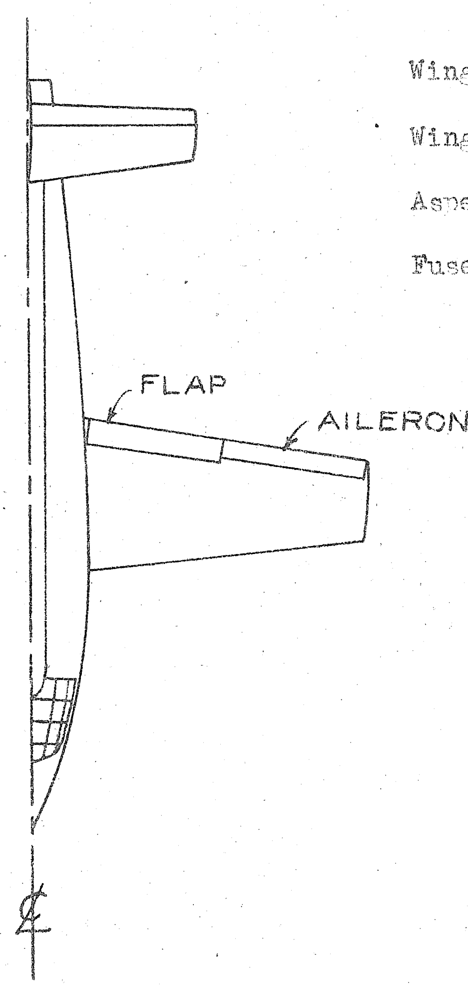
One of the most serious problems is the determination of the shearing stress distribution at the pointed leading and trailing edges under shearing flow resulting from both bending and torsion. Present day analysis of the stress concentrations in such a region is entirely inadequate for the calculation of an efficient design, and Professor Timoshenko of Stanford University has been given an Army Research contract to develop an improved method. Another difficult problem is the distribution of stresses in the triangular wing root region of a swept-back wing as shown:



This region cannot be analyzed by the present day practice of isolating each wing section because the problem is essentially a three-dimensional one. The California Institute of Technology is now working on this problem under Army contract. A third difficulty of plate construction is its inadaptability to cut-outs such as flaps and slots, especially in the region of the leading edge. The mechanical design of these auxiliary lift devices still remains a secondary but unsolved problem.

As would be expected, aerodynamic research has led structural research in the opening of the transonic field. The situation at the present time is that aerodynamic research has outlined the problems which are ahead, but much structural research must be done before an efficient design can be made. The two airplanes which have thus far been designed and built in America for flight research at transonic speeds are the Bell XS-1 and the Douglas D-558 (Skystreak), which were built under the auspices of the Army Air Forces and the NACA. A brief description of the wing design of each follows.

<u>Bell XS-1 Wing</u>



Wing Span - 28 ft.

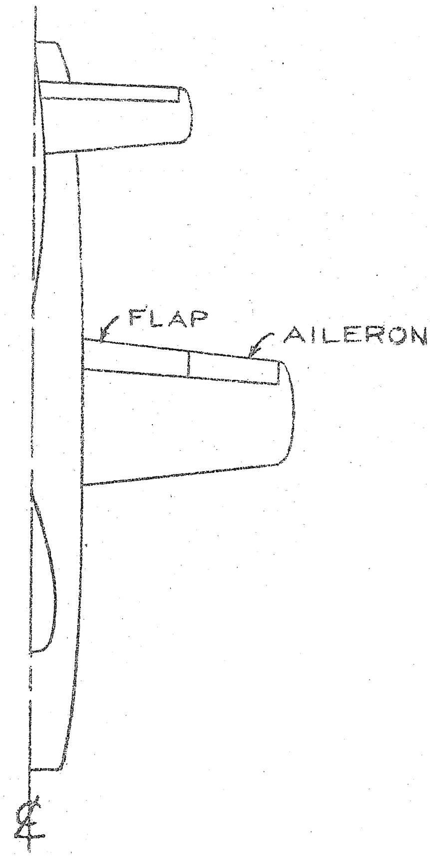
Wing Area - 130 sq. ft.

Aspect Ratio - 6.0

Fuselage Length - 31 ft.

As is seen in the plan view, the Bell XS-1 wing does not use sweep-back to raise the critical Mach number. It does, however, incorporate a thin low-drag airfoil of only 8% thickness at the wing root. Wing loading is 100 lbs./ft.^{2} at take off, and the wing is stressed for a design load factor of 18 g. Ordinary trailing edge flaps are used as auxiliary lift devices. A second transonic model, the XS-2, has been projected by Bell which will make use of a large amount of sweep-back, but this airplane has not yet been constructed. The wing construction is of a novel type, being full monocoque, machined from a single solid piece of aluminum alloy. Skin thickness tapers from 5/8" at the root to 1/8" at the tips. This type of design is very adaptable to changes in section, and would seem well suited to an experimental airplane. It may even be used extensively on production types if the construction cost can be kept down. It is not easily analyzed by present methods, and presents a worthwhile problem for future research. There is a possibility, however, that the curved low-drag airfoil may eventually be made obsolete by the flat plate diamond airfoil which is much more easily analyzed, except for the stress concentrations previously noted.

Douglas D-558 Wing



WING SPAN - 25' 0"

WING AREA - 150 sq. ft.

ASPECT RATIO - 4.2

FUSELAGE LENGTH - 35' 1"

The Douglas wing, like the Bell XS-1 wing, does not use sweep-back to raise the critical Mach number. A second model, the D-558, Phase II, is projected which will incorporate a large amount of sweep-back. Like the Bell XS-2, however, this plane has not yet been built. The Phase I has a thin low-drag airfoil of 10% thickness at the root, and a wing loading at take-off of about 65 lbs/ft^2. The wing is stressed for a design load factor of 18g. Ordinary trailing edge flaps are used as auxiliary lift devices. The structure of this wing is more nearly conventional than the XS-1, having a dural sheet skin supported by ribs and longitudinal stiffeners. The skin at the root is built up of three laminations riveted together, with a total thickness of about 1/2". Two of the laminations are successively dropped towards the tips, leaving a single layer of 0.060" skin at the tips. This laminated construction seems to be a makeshift solution necessitated by lack of facilities for handling heavier sheet thicknesses, and will probably be abandoned as better production methods and equipment become available. It would therefore seem pointless to devise special analysis techniques for this type of construction.

Part VII. Selected Bibliography

I. Swept Wings at High Speeds

1. Swept-back Wings at High Velocities. H. Ludwieg. Navy Translation CGD-19. (1946).

   Three-component wind tunnel tests on a series of 6 airfoils with sweep from -45° to $+45^{\circ}$ at Mach Numbers from 0.5 to 1.21. Shows experimentally the great reduction of drag at high subsonic Mach Number caused by sweep. Also shows the change in the slope of the stability curve, including the reversal of slope caused by forward sweep.

2. A Swept-Back Wing at High Speeds. Multhopp. Navy Translation CGD-34. (1945).

   A theoretical investigation of the criteria for raising the critical Mach Number by means of sweep-back. Calculations are also made concerning the oblique shock waves formed on a swept-back wing above the critical Mach Number, and of the air stream velocity over the surface of a swept-back wing. An investigation is made of the effect of the position of maximum thickness along the chord. Curves are presented showing the effect on the critical Mach Number of sweep, wing thickness, and position of maximum thickness. The undesirable moment characteristics of swept wings are examined.

3. Swept Wings at High Speeds. A. Busemann. Navy Translation CGD-84. (1946).

A lecture on transonic flow phenomena illustrated with wind tunnel data taken from swept wings up to a Mach Number of 1.2. This lecture is a very clear description of the physical nature of both subsonic and supersonic flow, and the mechanism of the drag reducing effect of sweep-back.

II. Swept Wings at Low Speeds

4. German Research on Swept-back Aerofoils. M. Gdaliahu. Royal Aircraft Establishment Tech. Note No. Aero. 1706. (1946).

This note contains a survey of German wing tunnel research programs on swept-back airfoils at low Mach numbers. It also contains a list of German reports with summaries, and a summary of the Multhopp and Weissinger methods for calculating spanwise lift distribution over a swept wing which were developed in Germany during the war.

5. Contributions to Sweep-back Research. H. Luetgebrune. Navy Translation CGD-128. (1946).

Very extensive wind tunnel tests to determine the performance of a swept wing, supplemented by considerable discussion. It is found that the flow over the outboard sections of a swept-back wing deteriorates due to excessive boundary layer buildup. For this reason, nose slots and flaps are much more effective than trailing edge flaps or ailerons, which lose a large amount of their effect. The boundary layer causes separation to occur first at the tips of a swept-back wing. The wind tunnel tests form a comprehensive program to determine the effects of sweepback on $dC_L/d\alpha$, C_L max, C_D, and dC_M/dC_L; of dihedral on $dC_L/d\alpha$, C_L max, and $dC_L/d\varphi$, (where φ is the angle of yaw); and of twist on $dC_L/d\alpha$, C_D, and C_L max. Extensive curves of experimental results are presented. The effects of the following auxiliary lift devices are also investigated: leading edge slots, rotable nose, end plates and fences, split flaps, cambered flaps, and slotted flaps.

6. Swept-back, Truncated Swept-back, and M-Shaped Wings. Lemme. Navy Translation CGD-36. (1945).

   Wind tunnel tests involving a large number of models in an attempt to determine how much the basic shape of a wing can be altered by sweep-back and still remain relatively free of the serious stability disadvantages of a swept wing. Several truncated swept-back wings are found which give relatively low drag and yet retain good stability characteristics. An investigation is also made of the effect of flap deflection on wing performance.

7. Preliminary Wind Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept-Back Wings. Wm. Letko and Alex Goodman. NACA Tech. Note 1046. (1946).

   Wind tunnel tests on wings with sweepback from 0° to 60° to investigate the effects of sweep on longitudinal stability, lateral stability, lift curve slope, and the effectiveness of ailerons, split flaps, and spoilers. It was found that ailerons, split flaps, and spoilers were ineffective on wings of large sweep, altho spoilers may be useful for directional control due to their large drag. It was also found that wing tip droop was beneficial to the stability characteristics.

8. Measurements for a Tapered, Swept-Back Wing. Th. Schwenk. Navy Translation CGD-38. (1945).

Six-component force tests and pressure distribution measurements in a wind tunnel on a wing of 32° sweep-back at a Reynold's number of 2.35×10^6. The main purpose of the study was to determine the effects of yawing a swept-back wing. Models were tested with and without slots. Effects of increasing yaw: C_L decreases, C_D increases, side force increases, and the rolling moment stability coefficient $dC_l/d\beta$ increases rapidly. Curves are given.

9. The Effect of Geometric Dihedral on the Aerodynamic Characteristics of a 40° Swept-Back Wing of Aspect Ratio 3. Bernard Maggin and Robert E. Shanks. NACA Tech. Note 1169. (1946).

Wind tunnel force tests were made at low Reynold's Numbers to determine the effect of geometric dihedral on the effective dihedral at various lift coefficients for a swept wing. It was found that strongly swept wings show extremely large effective angles of dihedral at high lift coefficients. This undesirable characteristic can be reduced somewhat by negative geometric dihedral, but at the cost of reducing the lift curve slope.

10. Increasing the Lift of a Swept-Back Wing. Brennecke. Navy Translation CGD-243. (1946).

Wind tunnel tests of attempts to raise C_L max of a 35° swept wing by means of slats and split flaps. Results show that split flaps were rather effective in increasing the lift. Slats had little effect on lift, but reduced undesirable stalling moments appreciably.

11. Wind Tunnel Tests of a 45° Swept-Back Wing with Nose Flaps. Krüger. Navy Translation CGD-242. (1946).

Preliminary wind tunnel tests on nose flaps. Not very good results due to unsuitable models used because of expediency. Does show, however, a decided increase in C_L max even for an airfoil (NACA 23018) with a round nose and large maximum thickness.

12. The Nose Flap as a Means for Increasing the Maximum Lift of High Speed Aeroplanes. W. Krüger. Navy Translation CGD-532. (1946).

Wind tunnel tests of wings equipped with nose flaps, both alone and in conjunction with split flaps. Results show great increase of C_L max of thin wings due to nose flaps, and the increase in effectiveness of split flaps when used in conjunction with nose flaps. The effect of the nose flap is to round the sharp nose of a high speed airfoil thus preventing separation as the angle of attack is increased. Curves are presented showing that the nose flap has an optimum deflection angle which must be attained quickly because wing lift with the flap partly deflected suffers greatly.

13. Investigation of the Wake and Drag of Straight and Swept-Back Wings. H. Luetgebrune. Navy Translation CGD-328. (1946).

Wind tunnel wake surveys which show a peculiar increase in C_D of a swept wing at high angles of attack in the neighborhood of Reynold's Number of 0.3×10^6. The drag increase was found to be a Reynold's Number friction effect. The report is valuable in that it contains considerable data on the wake pattern behind a swept-back wing.

14. Wake and Drag Tests on Swept and Straight Wings. H. Luetgebrune. Navy Translation CGD-106. (1945).

Wind tunnel wake surveys on wings of forward and back sweep to compute profile drag and separation effects. Results show that separation begins first at the tips of a swept-back wing and at the root of a swept-forward wing. Results also show a large increase in C_D for swept wings at certain Reynold's Numbers and high angle of attack which is due to friction effects.

III. Load Distribution over Swept Wings

15. Theoretical Distribution of Load over a Swept-Back Wing. Doris Cohen. NACA Wartime Report L-221, Originally ARR. (1942).

The theoretical distribution of load over an elliptical wing of 30° sweep-back is calculated by the vortex theory. Results show a shift of load from the center of the wing towards the tips. The method does not seem to be applicable to arbitrary plan forms.

16. Pressure Distribution Measurements of Swept-Back Wings (Second Report). H. Luetgebrune. Navy Translation CGD-133. (1941).

Wind tunnel tests on 3 wings: 35° forward sweep, straight, and 35° sweep-back. Pressure measurements are given showing the chordwise and spanwise lift distribution over the wing. Lift curve slope, pitching moment characteristics, and center of pressure location are also given. Comparison with the Multhopp lift distribution theory shows good agreement for the straight and swept-back wings, but poor check for the swept-forward wing. Shows that load moves towards the tips on a swept-back wing and towards the root for a swept-forward wing. Separation occurs at the point of maximum load in each case at about the same angle of attack as for the unswept wing. Lift curve slope was not materially affected by the amount of sweep in the wings tested.

17. The Lift Distribution of Swept-Back Wings. Weissinger. Navy Translation CGD-124.

Two theoretical methods of calculating the spanwise lift distribution for a swept wing by the circulation theory are given, one assuming a lifting line; the other a lifting surface. The methods are essentially an extension of Prandtl's work. A little work is also done on wings in sideslip. The lifting surface method is rather laborious, but the lifting line method seems very good.

18. Pressure Distribution Measurements on Arrow (Swept-Back) Wings of Constant Chord in Symmetrical Airflow. Jacobs. Navy Translation CGD-83. (1945).

Wind tunnel tests on 4 wings of sweep-back ranging from 0^o to 45^o at various angles of attack. The object of the tests is an experimental check of the theoretical lift distribution methods of Multhopp and Weissinger. Results do not check Multhopp's method very well, but give good agreement with Weissinger's Lifting Line Theory.

19. Pressure Distribution Measurements on Two Arrow Wings of Constant Chord (Sweep-back angle of 30^o and 45^o) for Asymmetric Airflow. Jacobs. Navy Translation CGD-240. (1944).

Wind tunnel pressure distribution measurements were made on the two wings at 6 angles of attack and angles of yaw from 0^o to 20^o. A theoretical method of predicting the lift distribution for swept wings in yaw which gives good agreement with the observed results.

IV. Unswept Wings at High Subsonic Speeds.

20. Fundamentally New Means of Increasing the Performance of High Speed Aircraft. Th. Zobel. Navy Translation CGD-107. (1945).

A qualitative but thorough analysis of the possibilities of increasing performance by the use of sweep-back, proper nacelle (jet engines) placement on the wing to avoid interference, and proper fuselage shape. Considerable attention is also paid to the possibility of reducing wing drag by boundary layer removal, using the jet engines as pumps. Quantitative test curves are given which show considerable drag reduction by adequate boundary layer removal.

21. Theory of the Airplane Wing in Compressible Media. L. Prandtl. Navy Translation CGD-110. (1936).

A theory investigation of compressible and supersonic flow by the use of velocity and acceleration potentials. This report is somewhat dated, but contains valuable basic material.

22. Tests on Airfoils Designed to Delay the Compressibility Burble. John Stack. NACA Tech. Note 976. (1944).

Wind tunnel force tests of the NACA 16 Series airfoils, with a discussion of the theories which lead to their design. These airfoils give low drag at high subsonic Mach Numbers because they maintain laminar flow over a large portion of the chord.

23. Velocity Distribution on Wing Sections of Arbitrary Shape in Compressible Potential Flow. L. Bers. NACA Tech. Note 1012. (1946).

A theoretical computation of the velocity distribution on a wing section of arbitrary shape in compressible flow, assuming the adiabatic equation of state holds rigorously. Comparison is made with experimental results showing close agreement. A computation method for practical use is presented, together with tables and charts to facilitate the work.

24. Effect of Mach Number and Reynold's Number on Maximum Lift Coefficient. J.R. Spreiter and P.J. Steffen. NACA Tech. Note 1044. (1946).

Flight test and wind tunnel data on 6 modern fighter type aircraft at Mach numbers from 0.15 to 0.72 and Reynold's numbers from 4.4×10^6 to 19.5×10^6. Results show that Mach number has considerable influence on the maximum lift coefficient, which decreased up to the critical Mach number, and then increased somewhat after that for the low-drag airfoils. Reynold's number had much less effect on the maximum lift coefficient, with practically no variation at all above a Mach number of about 0.55.

25. The Effect of Compressibility on Drag. S. Hoerner. Navy Translation CGD-518. (1946).

A comprehensive theoretical analysis of the effect of compressibility on friction and pressure drag of bodies of revolution and of wings. The critical Mach number for various bodies is also examined. All theoretical results are compared with experimental measurements.

26. Effects of Compressibility on the Maximum Lift Characteristics and Spanwise Load Distribution of a 12 foot Span Fighter Type Wing of NACA 230 Series Airfoil Sections. Pearson, Evans, and West. NACA Wartime Rep. L-51, Originally ACR-L5610. (1945).

Wind tunnel test up to a Mach Number of 0.70. Results show that maximum lift coefficient decreased above a Mach number of 0.30, but very slowly above a Mach number of 0.55.

In the high Mach number range, maximum lift coefficient occurred at an angle of attack somewhat above the stall. Lift distribution was only very slightly affected by increasing Mach number (straight wing), load moving towards the tips.

27. Critical Mach Number of Thin Airfoil Sections with Plain Flaps. Max A. Heaslet and Otway Pardee. NACA Wartime Rep. W-2, Originally ACR-6A30. (1946).

Wind tunnel test to determine critical Mach number as a function of lift coefficient for several thin and moderately thick NACA low-drag airfoils. Results show that the lift coefficient can be effectively increased at relatively high Mach numbers with plain flaps without serious effect on the critical Mach number.

28. Effect of Compressibility on the Pressure Distribution Over an Airfoil with a Slotted Frise Aileron. Arvo A. Luoma. NACA Wartime Rep. L-266, Originally ACR-L4612. (1944).

Wind tunnel determination of the complete pressure distribution over a P-47B model airfoil with a slotted Frise aileron at Mach numbers from 0.25 to 0.76, for various angles of attack and aileron deflection. Results show the striking way in which aileron effectiveness decreases at high Mach numbers.

29. Correlation of Flight Data on Limit Pressure Coefficients and Their Relation to High-Speed Burbling and Critical Tail Loads. R.V. Rhode. NACA Wartime Rep. L-269, Originally ACR L4127. (1944).

Results of flight test data on a P-47C airplane show that limit pressure coefficient is a function of Mach number for Mach numbers above 0.3. Results are extrapolated by theory to above the critical Mach number and a working chart for determination of the pressure distribution and lift coefficient about the region of potential flow (region of compressibility burble) is given. The effect of skin wrinkles on burble is also considered.

30. Investigation of the Stability of the Laminar Boundary Layer in a Compressible Fluid. Lester Lees and C.C. Lin. NACA Tech. Note 1115. (1946).

A theoretical investigation of the stability of a two-dimensional laminar boundary layer by the method of small perturbations. Quite extensive and very technical.

31. High-Speed Investigation of Skin Wrinkles on Two NACA Airfoils. H. Robinson. NACA Tech. Note 1121. (1946).

Wind tunnel test of the NACA 66, 1-115 and 23015 sections with simulated skin wrinkles on the upper surface near the nose. Tests were run up to a Mach number of 0.73. Results showed the wrinkles had little effect on C_L and C_M, but raised C_D to some extent and lowered the critical Mach number.

32. Summary of the Drag Characteristics of Practical-Construction Wing Sections. John H. Quinn. NACA Tech. Note 1151.

Wind tunnel and flight test investigations of the effects of manufacturing imperfections on the drag coefficients of actual wings at high Reynolds numbers. Results show that minor imperfections such as spar joints can cause an increase in drag of as much as 100% if they occur in the laminar flow region of low-drag wings. It was also found that wrinkling under load increased no-load drag by as much as 33%. De-icers were also very detrimental to drag performance. In each case, the main effect of the imperfection was to speed transition from laminar to turbulent flow. It was found that wings with spars placed aft of the laminar flow region and a heavy skin with a glazed surface give results approximating those of a perfectly fair surface.

33. High Velocity Measurements for a Wing of a Small Aspect Ratio. B. Gothert. Navy Translation CGD-37. (1945).

Wind tunnel tests on wings of a 1.15 aspect ratio at Mach numbers from 0.3 to 0.9. Results show that the lift coefficient does not obey the Prandtl $(1/\sqrt{1 - M^2})$ rule in wings of small aspect ratio, due to tip effects. The steep drag rise at the critical Mach number is also delayed, in fact, the critical Mach number of a wing of 1.15 aspect ratio is as high as that of a wing with an aspect ratio of 6 and 30° sweep-back. Lift curve slope is almost constant throughout the test range. Schlieren investigations show that separation does not occur even at a Mach number of 0.9 and an angle of attack of 8°.

34. The Tailles Construction of the Turbo-Jet Fighter Compared to the Standard Model. Kappus. Navy Translation CGD-324. (1946).

A comparison of a projected tailless fighter aircraft with the ME-262. The paper is a very good analysis of the various problems encountered in the aerodynamic design of a high speed aircraft and the methods by which they may be solved.

35. Wind Tunnel Wall Corrections in Compressible Flow. Wiesselsberger and Hemke. Air Technical Service Command, T-2 Intelligence Report No. 102. (1945).

Tunnel wall corrections for compressible, subsonic flow are computed both in regions where the Prandtl Wing Theory is applicable and where it is not. Results are given for close and open throat round tunnels.

V. <u>Airflow Above Mach Number 1</u>

36. Problems of High Speed Flight. James H. Bartlett. Army Air Forces Translation F-IR-2-RE. (1946).

This is a very useful report for background material. Its purpose is essentially the same as this work: to evaluate the present state of knowledge in high speed flight, and to present a working bibliography of reports. It also contains a number of interviews with leading German scientists. Many of the reports mentioned are in German.

37. Two Dimensional Irrotational Mixed Subsonic and Supersonic Flow of a Compressible Fluid and the Upper Critical Mach Number. Tsien and Kuo. NACA Tech. Note 995. (1946).

A very extensive theoretical investigation of two dimensional irrotational mixed subsonic and supersonic flow of a compressible fluid by hodograph methods.

38. A Condition of the Initial Shock. Th. Theodorsen. NACA Tech. Note 1029. (1946).

A theoretical investigation of the critical conditions for the formation of a shock wave in the transonic region of a wing. Results show that the shock wave is preferred which will reduce the local Mach number to 1.

39. The Interaction Between Boundary Layer and Shock Waves in Transonic Flow. Hans Liepmann. Journal Aero. Sciences, Vol. 13, No. 12, Dec. 1946, pp. 623-637.

Wing tunnel experiments of transonic flow on circular arc profiles show that state of the boundary layer affects the shock wave pattern. Shock waves can reflect off a boundary layer in a manner similar to reflection off a free jet boundary.

40. Upon the Position of Shock Waves on Profiles in a Flow. A. Betz. Navy Translation CGD-604. (1946).

A qualitative discussion of the position of a shock wave on a body in transonic flow, as affected by Mach number and by boundary layer. Increased Mach number causes a shift to the rear; increased boundary layer thickness causes a shift to the leading edge. It is found that boundary layer separation and the shock wave seriously affect each other, and that the exact flow is very difficult to evaluate.

41. A Study of Compression Shocks with Boundary Layer Separation. Eggink. AAF Translation F-TS-1026-RE. (1946).

It is known that compression shock waves transform supersonic flow to subsonic, and at the same time cause separation of the boundary layer. The nature of these shock waves is investigated for bodies of revolution and for airfoil sections both by Schlieren apparatus and by theoretical analysis, and a theory of shock branching is developed. A detailed shock polar diagram for branching shocks is also given.

42. Aerodynamic Lift at Supersonic Speeds. A. Busemann. Navy Translation CGD-111.

A theoretical analysis of supersonic flow. The paper, which was originally published in 1936, is somewhat out-dated, but contains a very good basic analysis of the nature of supersonic flow.

43. Two-Dimensional Wing Theory in the Supersonic Range. H. Hönl. AAF Translation F-TS-916-RE. (1947).

A theoretical investigation of two-dimensional supersonic flow about a plane vibrating wing by use of an acceleration potential method within linearized theory. It is essentially an extension of the Prandtl wing theory to the non-stationary case, resulting in an integral-differential equation which is solved by Laplace transforms. A good analysis of non-steady supersonic flow.

44. Introduction to the Aerodynamics of a Compressible Fluid. H. W. Liepmann and A.E. Puckett. Published by John Wiley & Sons. (1947).

A textbook on subsonic, transonic, and supersonic aerodynamics which provides an excellent basic understanding of compressible flow.

VI. Structures of High Speed Wings

45. The Frequencies of Cantilever Wings in Beam and Torsional Vibrations. C. P. Burgess. NACA Tech. Note 746. (1940).

    Development of a method for calculating beam and torsional vibration frequencies by energy method approximations. Method takes into account any spanwise variation of section strength and weight, and can be applied to any desired degree of accuracy by successive approximations.

46. Bending-Torsion Flutter Calculations Modified by Subsonic Compressibility Corrections. I.E. Garrick. NACA Tech. Note 1034. (1946).

    Theoretical investigation of the effect of subsonic compressibility corrections to flutter frequencies. Equations are given for the solution of any general problem. The effect of compressibility is generally small, but in high density sections may be as much as 17%.

47. Normal Pressure Tests on Unstiffened Flat Plates. R.M. Head and E. E. Sechler. NACA Tech. Note 943. (1944).

    Results of experimental tests on numerous plates of 0.010" to 0.080" thickness and 10" x 10" to 10" x 40" area. Deflection pattern under normal pressure was determined. A good source of plate deflection data.

48. Clamped Long Rectangular Plate under Combined Axial Load and Normal Pressure. Woolley, Corrick, and Levy. NACA Tech. Note 1047. (1946).

    A theoretical investigation of plates under normal pressure with clamped edges. Results show that normal pressure has less effect than on plates with simply supported edges, and that neglecting the effect of pressure is conservative in the design of the plate.

49. Buckling Stresses of Simply Supported Rectangular Flat Plates in Shear. Manuel Stein and John Neff. NACA Tech. Note 1222. (1947).

A more accurate and general extension of Timoshenko's plate formulas for shear to cover the cases of anti-symmetrical buckling. Curves are presented which give the plate buckling constant used in Timoshenko's standard formula.

50. Effect of Normal Pressure on the Critical Compression and Shear Stress of a Curved Sheet. Norman Rafel and Charles Sandlin, Jr. NACA Wartime Rep. L-57, Originally ARR-L5B10. (1945).

Experimental compression test on curved sheet specimens subjected to normal pressure. Results show that the effect of pressure is to raise the critical compressive stress and shear stress when the buckling is toward the concave side of the curve (the side under pressure in these tests) and to lower the critical stress when buckling is toward the convex side. The effect of normal pressure is quite appreciable.

51. The Shearing Rigidity of Curved Panels under Compression. N. J. Hoff and Bruno A. Boley. NACA Tech. Note 1090. (1946).

Experimental test to determine the shearing rigidity of curved panels under compression, especially in the buckled state. The panels are curved sheets stiffened by stringers and rings. Results show the shearing rigidity decreases greatly under compression load. The results are presented in graphical form, and empirical formulas are determined.

52. An Empirical Formula for the Critical Shear Stress of Curved Sheets. Paul Kuhn and L.R. Levin. NACA Wartime Rep. L-58, Originally ARR-L5A05. (1945).

The development of an empirical formula for the critical shear stress of curved sheets from experimental tests. The formula applies for sheets of r/t greater than 300, less than 1 radian of arc, and arc length/axial length less than 1.

53. Description of the ME 163 B-0 Construction. T-2 Intelligence Report No. 502.

A very detailed description of the construction of the ME 163 B-0, a German high speed turbojet fighter, including performance data. Reviews many problems which have been solved in the design of the plane.

54. ME 163 B Service Manual, Vol. I and II. Navy Translation CGD-255 and 256.

The very complete service manual of a German high speed fighter with swept-back wings (15°). A wealth of data on this design.

Part VIII. Supplementary Bibliography

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56. High Speed Measurements on a Swept-Back Wing. (Sweep-Back Angle 35°). B. Göthert. NACA Tech. Mem. 1102. (1947).

57. High Speed Measurements on a ME 163-B Tailless Airplane Model with Non-Twisted Wing. Lindemann. Navy Translation CGD-329. (1946).

58. Drag of Two Swept Wings with and without Fuselage and Nacelle at Zero Lift in the Region of Subsonic Compressible Flow. Straasl. Navy Translation CGD-513. (1942).

59. Pressure Distribution Measurements at High Speed and Oblique Incidence of Flow. A. Lippisch and W. Beuschausen. NACA Tech. Mem. 1115. (1947).

60. Three and Six Component Measurements of a Model of the ME 329 with Swept-Back Wings. Hildenbrand and Lemme. Navy Translation CGD-102. (1946).

61. Effect of Reflex Camber on the Aerodynamic Characteristics of a Highly Tapered Moderately Swept-Back Wing at Reynolds Numbers up to 8,000,000. W. Conner. NACA Tech. Note 1212. (1947).

62. Effect of Sweep-Back and Aspect Ratio on Longitudinal Stability Characteristics of Wings at Low Speeds. J.A. Shortal and B. Maggin. NACA Tech. Note 1093. (1946).

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68. The Calculation of Span Load Distribution of Swept-Back Wings. William Mutterperl. NACA Tech. Note 834. (1941).

II. <u>General High Mach Number Subsonic Flow</u>

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70. Suggestions Concerning Questions Submitted by Mr. L.E. Root About Experiences with High Speed. S. Gunter. Navy Translation CGD-42. (1945).

71. The Increase of Drag of Profiles in the Range of High Subsonic Speeds. B. Göthert. Navy Translation CGD-484. (1946).

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