THE DESIGN OF TAILLESS AIRPLANES

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SUMMARY

This paper reviews the various sources of information pertaining to the design of tailless airplanes. The particular topic of any report or memorandum is illustrated by its application to the problems encountered in the design of a given airplane. The subjects mentioned are the determination of the sweepback angle, aspect ratio, taper ratio, twist, dihedral, airfoil section, spanwise loading, and aerodynamic center of the wing; the control elements; and the fuselage and fillet for high speed aircraft.

To illustrate the application of the source material the design of a tailless supersonic airplane is considered. The slow speed wind tunnel tests of the 1/12-scale model are included in the report.

INTRODUCTION

The problem of basic airplane design has received considerable attention in the last several years. This problem concerns the basic design of the airplane configuration as a whole, the relation of wing, fuselage, and empennage to one another, their purpose, practicability, and necessity. The focusing of attention on this problem has highlighted the work of several men, such as Alexander Lippisch of Germany and John Northrop of this country, who have felt that the standard airplane configuration was not necessarily the best. These men and others have considered basic designs dissimilar to the conventional airplane.

The development of the modern airplane has been an evolutionary process. The first designers-aviators built a multitude of types of airplanes, only a few of which were successful. They continued on the path of design suggested by the successful airplane, but after a few attempts, discarded the unsuccessful designs. Of necessity only the simplest and most stable craft would fly, due to the lack of knowledge of nearly all aircraft theory and practice by the designers. The fact that the most stable airplane had a straight wing, and a tail at some distance from the wing, led to the improvement of airplanes of this type and eventually the subsidence of all other types. Thus,the basic design of the modern airplane was evolved, its existence being due to the fact that it was originally more stable and could thus be flown more easily than any of its competitors in design.

In the ensuing years more persons were considering the problem of most efficient airplane performance and their conclusions were that a higher performance flying machine could be achieved if the body and empennage could be discarded, leaving only the wing for lift and stowage space and the propulsive unit for flight. At first the advocates of this type of design thought only in terms of reducing the parasite drag of the airplane by eliminating the fuselage. This was the basis for their efforts until World War II and the appearance of the very high speed pursuit airplanes. Theory had indicated, and experience was proving, that the drag of conventional airplanes increased tremendously at speeds near the speed of sound. Also, at very high Mach numbers the airplane was apt to become unstable. Faced with the fact that the standard design was no longer clearly superior, a few designers considered modifications. In 1936, Busemann suggested sweeping the wing as a means of increasing the critical speed. In this country, Robert Jones made a similar suggestion in 1944. But since sweeping the wing made it possible to use the wing itself for the entire control of flight, the next step was to discard the tail. Hence, the satisfactory design of a tailless airplane became a necessity, from the standpoint of more efficient performance for large, relatively slow airplanes, and for transonic and supersonic aircraft and missiles.

NOMENCLATURE

b	Wing span
S	Wing area
AR	Aspect ratio
c	Wind chord, parallel to axis of symmetry except where noted
τ	Taper ratio, ratio of tip chord to root chord
Λ	Angle of sweep of wings, measured at the line connecting the 25% chord points of the wing sections, except where noted
M	Mach number, ratio of speed of airplane to the free stream speed of sound
a.c.	Aerodynamic center
c.p.	Center of pressure

PROBLEMS OF TAILLESS AIRPLANES

Consider the outward appearance of the conventional modern airplanes. Whether they be large or small, slow or fast, they all have the same general shape, differing only in number of engines and vertical tails and location of the wing on the fuselage. Not so the tailless airplane! In the latter type the aspect ratio, taper ratio, sweepback angle, and size of fuselage all depend on the primary purpose of the airplane in which they are incorporated. G. H. Lee, in reference 1, gives a broad outline of the basic types of tailless aircraft, and the type of application for which each case is suited. He concludes that large, heavy airplanes should have the fuselage entirely submerged in the wing, while the sweepback angle would be determined by stability considerations. The high speed airplane should have a fuselage plus a wing large enough to avoid high lift coefficients at cruising speeds, while the sweepback angle of the wing is determined by both drag and stability considerations.

From the brief description of the problems of the design of tailless airplanes, one fact becomes outstanding: at the present time there is no broad background of practical experience to refer to in the design of this type of airplane. This will probably be the case for some time to come, because of the fact that the appearance of the airplane will depend so much on the application for which it is intended. Therefore, in the following procedure for the design of tailless aircraft, with the references given for the various phases of the design, one must be aware that the specific problems encountered in a given airplane will be so important as to dictate the method or procedure in the design of that airplane.

Furthermore the references given will in many cases not apply to extreme types of design. This is especially true for very small high-speed airplanes with extremely large leading edge sweepback angles, and relatively large bodies, compared to the wing span.

In the following part of the report reference will be made to the specific problems of two types of airplanes: the slow speed type with relatively little sweep to the wing, and the very fast subsonic, or supersonic, aircraft with large amounts of sweepback. In a majority of cases the airplanes to be designed will lie somewhere between these extremes, and the problems will be different, but the designer who is aware of the limiting conditions has a basis for an approximation to the solution.

WING DESIGN

It seems reasonable to expect that the wing of a tailless airplane will entail more design work than the wing of a conventional airplane, due to the larger number of variables and uses of the wing. That this is actually the case can be seen from the following procedure to establish the wing planform and aerodynamic characteristics.

The two basic parameters of airplane performance and design are the gross weight and maximum velocity. Though the decision as to these two factors might be influenced by some other criterion, in general they control the design of the rest of the flying wing airplane, reference 1.

For a given airplane configuration, such as fighter, light bomber, or long range bomber, the gross weight can be approximated from experience with conventional airplanes that perform the same function, or else a weight breakdown can be attempted to estimate the weight. The airplane top speed can be chosen, thus determining the power plant to overcome the estimated drag, or else the airplane will be built for a given engine, and the speed will then depend on the drag. In either case the drag coefficient, C_D, and the wing area, S, will have to be estimated in order to find the drag.

The wing area can be found fairly closely by using wing loadings of comparable conventional airplanes. The wing loading itself must be given some thought because it can have an effect on lateral stability, reference 2. The wing area is also a function of the allowable landing speed, since swept-back wings inherently have lower maximum lift coefficients than straight wings. Thus the wing area is a compromise between a desire for a high maximum speed and a low stalling speed. But since the sweepback angle is not yet known, it is better to make a preliminary estimate of wing area, and then try to estimate the approximate drag coefficient.

At the present time there is a somewhat limited amount of experimental data on low speed drag and moment characteristics of swept-back wings for nearly all types of wing planforms, but the high speed subsonic, transonic, and supersonic data is extremely scarce. This means that the estimate of the drag coefficient will depend to a great extent on theory for high speed flight.

For low speed airplanes the drag coefficient can be estimated from wing alone tests on conventional airplanes, for tailless aircraft with low sweepback, or from the experimental data of references 3, 4, 5, 6, and 7. In many cases it is necessary to know the approximate angle of sweepback even for low speed tests and it is mandatory for high speed drag calculations. Robert Jones, in references 8 and 9, clearly shows that the theoretical condition for supersonic flight with straight swept-back wings is that the component of velocity perpendicular to the wing panel axis be less than the speed of sound. In actual practice the component of velocity must be less than the critical velocity of the wing section.

At this point there are two possibilities, depending on the design procedure. If a definite top speed is desired the sweepback angle of the wing can quickly be estimated from the above sources, using references 10 and 11 to find the critical Mach number of a wing at high subsonic flight. In the more usual case where the drag will determine the top speed it must be assumed that the wing will be swept back to an angle sufficient to appreciably diminish the effect of wave drag, which can be approximated from the two-dimensional theory given in references **12** and **13**, and from the three-dimensional theory of references **14**, **15**, **16**, **17**, and **18**. The fuselage drag coefficient, which is discussed in the following pages, is added to the wing coefficient to get the total drag coefficient. Knowing the overall preliminary drag coefficient, the maximum velocity can quickly be estimated, and from this preliminary velocity, the induced drag coefficient of the airplane can be approximated from the references already given for the wing drag. The estimated maximum velocity can now be determined and thus the necessary sweepback angle of the wing can be found from the references given previously.

It must be remembered that the drag estimate given above was made solely to determine the sweepback angle of the wings, before the aspect ratio, taper ratio, or other wings characteristics could be determined. Thus, when the two other major items in the wing design, the aspect ratio and taper ratio, are found, the drag and velocity estimates must be reviewed to see if they have been changed because of better knowledge of the airplane layout. In other words the design of any airplane is a repetitive process in that certain arbitrary assumptions must be made in order to determine factors that can be used to check the initial assumptions. For the airplane in question the three parameters found so far are gross weight, wing area, and sweep angle of the wing.

Up to this point mention of the sweep angle of the wings has been in terms of sweepback, but this is not the only case since theoretically sweep forward will work as well as sweepback. The advantages of sweep forward over sweepback are given in references 2 and 19. The principal advantage is that wing-root stalling occurs before the wing tips stall, while the opposite effect is observed for swept back wings. The disadvantage of forward swept wings is that the forward travel of the center of pressure requires the fuselage to be very far forward. Not much more can be said about this configuration because at the present time nearly all the data applies to swept back wings, the Germans being the only ones who have made any tests at all on the other types.

The other factor that has not been clearly defined in the above discussion of the necessary sweepback angle is the reference line to which the sweepback is measured. This again depends upon the type of airplane to be considered. For the relatively slow speed airplane with little sweep to the wings the angle of sweep is usually referred to the line connecting the 25% points of the wing sections. This is done because the section aerodynamic centers are located approximately at the 25% point for small angles of sweep. The idea of sweep referred to the 25% line loses its meaning if carried to wings with fairly high sweepback angles, of the order of 20° or greater, since for larger angles the section aerodynamic center moves rearward, reference 20. Actually, for supersonic aeroplanes the critical sweep angle can be considered to be either the leading edge or the line connecting the points of maximum thickness, as shown by Puckett in reference 15. At the present time the test data refers to the angle of the 25% line in all cases except as noted.

The aspect ratio is the next major item to be found in the determination of the wing characteristics. Systematic stability tests have been carried out to determine the effect of aspect ratio and sweep angle. These tests are shown in references 2, 4, 5, 6, 19, and 21. The results from isolated cases can be seen in references 3, 7, and 22. The results of all these tests indicate that the aspect ratio must decrease as the sweepback angle increases in order to maintain longitudinal stability at low speeds. The curve of aspect ratio versus sweepback angle, figure 1, is representative of the data from references 4, 5, and 6. It can be seen from this curve that airplanes with high sweepback angles must have a very low aspect ratio, which can be determined quite closely from the curve of figure 1. As can be seen from the data of reference 5 it is possible to use slightly higher aspect ratios for a given sweep angle if measures are taken to control the tip stall characteristics.

For stability reasons the taper ratio of highly swept wings should be from 0.6 to 0.3 since the data of reference 23 indicates that higher taper ratios greatly enhance the possibilities of tip stalling. Unfortunately this condition for the design of the wing tip does not agree too well with the theory for the design of supersonic airplanes given in the previous references, for theory indicates that a wing with pointed tips is best. The decision as to the proper taper ratio to use will depend on which condition is more important, high speed or stability at low speeds.

Summing up the conclusions arrived at so far it is seen that the gross weight, power, wing area, and sweepback angle, aspect ratio, and taper ratio have been approximated for high-speed airplanes designed to fly at supersonic or very high subsonic speeds. From this data the complete wing planform can be found, namely the span, root chord, tip chord, angle of sweepback of leading edge, etc. Having for the first time a clear picture of the wing shape, it is necessary to refer back to the references given on wing drag and check the calculations in the light of better knowledge. Thus, if it is known that a delta wing, or any wing with zero taper ratio, is to be used, then references 8, 9, 15, 16, 17, 24, and especially reference 18, can be used to make a more intelligent estimate of the total wing drag. If wings with cut-off tips are used, the above references should be used in conjunction with references 13 and 14, though the theory is not yet known for such wing shapes.

In the preceeding pages the sweepback angle and aspect ratio of the wing was considered for high speed aircraft, with the primary emphasis on speed and secondly on stability. For the large relativel slow-speed airplanes the situation is altered in that the drag considerations will not determine the sweep angle because stability and load carrying ability are the primary factors.

One fact that can be seen in all the low speed tests is that the basic maximum lift coefficient decreases with sweep, while the drag increases slightly for large sweep angles. Thus, the maximum lift over drag ratios will decrease with increase in the sweepback angle. Another effect is that the increment in lift coefficient attainable with a given flap will decrease with the sweep angle, while the angle of attack required for a given value of the lift coefficient will increase with the sweep angle. From these facts it would seem that a very low sweepback angle, if any, should be used, but the opposite effect is indicated in reference 29 where it is shown that "sweepback gives the wing an effective tail length adaptable for tailless airplanes, permits the use of high lift flaps at the center of the wing where their lift increments produce only minor changes in the pitching moment about the center of gravity of the airplane, allows flaps for longitudinal control to be located near the wing tips where only minor changes in lift are necessary to produce the requisite pitching moments for trim, and permits more leeway in locating the center of gravity inasmuch as the aerodynamic center of the wing can be controlled by the angle of sweepback."

The above discussion has perhaps helped make it clear that for low speed airplanes the determination of the sweepback angle can be quite difficult, since it will have an effect upon the maximum lift coefficient attainable with and without flaps, the landing angle, pitching moment, rolling moment, and several other factors in dynamic stability. The one bright light in the wilderness of indecision lies in the fact that in addition to the material on swept back wings, the many studies on straight wings with varying taper ratios and aspect ratios can be used to indicate the effect of modifications. Thus, the actual test data for wings with fairly low angles of sweepback is much more voluminous, and covers more variables, than does the data for highly swept back wings. At this time the best material for determining the correct wing shape can be found in references 2, 5, 6, 19, and 25. Other material applicable to special wing shapes, special problems, or as general reference material, can be found in references 3, 4, 21, 22, 26, 27, and 29.

In the above discussion no mention has been made of the procedure to be followed in the design of a slow speed tailless airplane. It is felt that if any procedure is to be established, it must of necessity be extremely indefinite due to the large influence of the type and application of any airplane under consideration, but an attempt will be made to set up the basic steps in design as follows: For a given gross weight the wing loading can be estimated from conventional values or from figure 16 of reference 2. For an arbitrary selection of an aspect ratio and taper ratio, the wing span can be calculated and the sweep angle can be approximated from the stability range curve shown in figure 1. The span and sweepback angle will give the effective tail length and thus the trimming moment of ailerons can be calculated, using the data of references 2, 6, and 20. This moment must be sufficient to overcome the adverse diving moment due to flaps, references 2, 6, 20, 26, and 29. The design of the ailerons and flaps, above, will be a function of the necessary lift and stability coefficients, which in turn will depend on the aspect ratio and sweepback angle. If the moment due to flap deflection could not be taken out by the ailerons, either the flaps could be reduced, the ailerons enlarged, or the sweep angle increased. It can now be seen that the various wing elements must be varied in a systematic order, if possible, in order to get the most efficient wing to perform a given function.

It is obvious that the above method is not precise, but until a great deal more systematic design data is made available it is the only possible procedure.

This paper has so far covered all parts of the basic wing design except for twist and dihedral. These are also a function of the purpose of the airplane, as, for instance, the twist of a wing designed for very high speed flight should be zero in order to minimize the drag, while the same wing should have washout in order to help prevent tip stalling at high lift coefficients. In general the good effect of twist is not enough to offset the adverse drag effect at high speeds, but it can be useful for low speed airplanes, as shown in references 19, 21, 23, and 28. The dihedral of a swept back wing affects the lateral stability and control to a great extent, as shown in references 2, 28 and 30. Since a swept-back wing has an "effective dihedral" due to sweepback, the actual dihedral angle can vary from positive to negative, depending on sweep angle, to get the correct rolling moment coefficient for the airplane.

The final element to determine for the basic wing is the aerodynamic section to be used. It must be remembered that in swept-back wings the section considered is that perpendicular to the line connecting the 25% points. This definition loses its effectiveness, as was previously explained, when applied to highly swept-back and tapered wings, so for these wings the section perpendicular to the line of maximum thickness would be more apt for design purposes.

The wing section for a high speed plane will be considered first. The requirements are that the drag be low for high Mach numbers, the critical Mach number high, the maximum lift coefficient high, and the pitching moment curve stable. All this in addition to having as much space as possible in the wing to hold fuel, control lines, etc. Since all these conditions cannot be met at the same time, it is necessary to amend them in view of the desired performance of the airplane. They can be evaluated as follows: a very thin wing, of the order of 5% t/c, will have low drag and a high crit Mach number, but the maximum lift coefficient will be of the order of 0.7 to 1.2 depending on the shape of the section. Furthermore, these thin sections will have poor stalling characteristics on wings and will certainly not afford much stowage space. A selection of a wing section will depend on the aerodynamic problems listed above, and the structural problems involved in manufacturing very thin wings. The question of stress determining the thickness will appear in the discussion of spanwise lift distribution.

The airfoil for a large airplane of slow speed need not be designed so much for drag as for lift and stability characteristics. Furthermore, in large airplanes it is advisable to consider fairly thick sections so that the airplane can be designed as a flying wing where all equipment is carried in the wing and the fuselage is eliminated entirely. Also in thick wing airfoils it is much easier to accommodate the various high lift devices listed in a subsequent section. Thus,the selection of any airfoil will be a function of the desired speed and load carrying capacity of the airplane, and also, to a certain extent, upon the actual size. In any event the selection can be based on the standard compilations of airfoil data.

In either high speed or slow speed airplanes a critical factor is the spanwise lift distribution. From the material given in the references it is obvious that many of the problems typical of swept-back wings occur because of the peculiar lift distribution. The general effect of sweepback is to increase the lift at the wing tips and reduce it at the root, while the opposite effect is observed with sweep forward. Thus it is seen that the high lift at the tips is a primary factor in premature tip stalling of swept-back wings of reasonable aspect ratio, which is one of the main reasons why wings with sweep forward are now being considered for tailless airplanes.

There are at present several different methods available for estimating the spanwise lift distribution of swept-back wings. The easiest methods, and those which give a very close approximation to the actual lift distribution, are applicable only for relatively slig angles of sweep, of the order of 15° to 20° at the most, while larger angles entail more computations of which the results are still not to good. The problem arises from the fact that the Prandtl lifting line theory cannot be truly applied to wings which are not straight, therefore either approximations must be made to the lifting line theory or else other methods developed, such as the lifting surface theory described in reference 31. In this paper Weissinger notes that two methods are available. These are called the F-method, or lifting surface method, and the L-method, or lifting line method. In either case account is taken of the distribution of the circulation over an area. The F-method, when applied to swept-back wings, requires about eight hours of computations while the L-method, which gives almost as good results, can be calculated in about one third the time. Experimental data from reference 20 indicates that the L-method gives fairly close agreement to the actual lift distribution.

Other methods for finding the lift distribution of swept wings can be found in references 32 and 33, while reference 27 can be extended to calculate the effect of small angles of sweep on the lift distribution.

An estimate of the spanwise lift distribution for the wing of the airplane shown later in this paper, $\Lambda = 63^{\circ}$ is given on figure 2, where the distribution is calculated from reference 27 and approximated from the curves given in references 32 and 33. From examination of the data of references 33 and 31 it appeared that the two methods gave much the same type of lift; therefore, figure 2 represents all the different methods. In each case the section lift coefficient corresponds to a total wing lift coefficient equal to unity, so the curves are directly comparable. Elliptic lift distribution is also plotted since Jones, in reference 8 shows that an elliptic lift distribution is approached for delta wings lying well inside the Mach cone. The lift distribution with flaps or ailerons deflected can be approximated from reference 29.

The practical application of the lift distribution curve comes in finding the stresses on the wing and the position of the center of pressure of the entire wing behind the feference point on the wing root chord. Thus, if it is known that the center of pressure of each section lies at the 25% chord point, the center of pressure of the wing must be referred to a point 25% of the root chord behind the wing apex. It is seen that this determination of the c.p. of the airplane is dependent on the wing characteristics.

For wings with no sweep the c.p. of any section will be at approximately the 25% point for $M < 1$, and at the 50% point for $M > 1$, references 13, 14, 16, 18, and 24. For delta wings of small vertex angle the c.p. theoretically remains at the center of the wing area, 2/3 of the wing root chord from the vertex, for all Mach numbers references 17, 18, and 24. For wing planforms in between the two extremes the c.p. will have values depending on both planform and Mach number.

DESIGN OF CONTROL ELEMENTS

In the first part of this report considerable mention has been made of the very high degree of importance the problem of stability and control assumes in the design of tailless airplanes. These problems arise from the single fact that the horizontal tail is not used, but instead the wing performs all control functions through the use of sweepback and special control devices. That sweepback is not an inherent part of the static longitudinal stability of wings is a well known fact, but if straight wings with reflex airfoils are used, the other aerodynamic qualities of the airplane are adversely affected. The different aspects of this problem are shown by Jones, reference 25.

Since the design of control elements for flying wings is a complete problem in itself no attempt will be made to discuss the design of the controls or the calculations and tests made to determine the stability coefficient derivatives. Instead the reader is referred to reference 19, which is the most complete and thorough source available at the present time. In this report the items discussed are: remedies for tip stalling through the use of wing twist, change in airfoil section, flat plate separators, changes in planform at the wing tip, leading-edge slats, and taper; effects of power on longitudinal stability; longitudinal control through the use of bevels, special venting, slots ahead of the elevators, automatically controlle tabs, or spoilers; lateral stability; directional stability of propellers, fuselages, fins, turned down wing tips, and automatic control; directional control; aileron control, using spoilers or elevons;

and dynamic stability. Many of the items listed above are quite thoroughly discussed in reference 2, while various stability problems are examined in references 5, 6, 21, 25, 34, 35, and 36. Aileron effectiveness for two dimensional airfoils at supersonic speeds is given in reference 13.

High lift devices are included with control elements because they have many problems in common, and in some cases are actually the same element performing two different functions, as for instance the outboard flap which can also be used for aileron or rudder control. Furthermore high lift devices are an inherent factor in stability considerations since they cause many of the stability problems and are also used to help solve some of them, as does boundary layer control on the wing tips increase the lift and also helps control the tip stall, which affects longitudinal and lateral stability.

The basic type of lift increasing device is the trailing edge flap, the analysis of which is given in reference 35. Various other lift devices are given in references 37 and 38. These variations are usually modified wing leading edges, such as a drooped nose, slot, or leading edge slat. An additional means of increasing the lift is through boundary layer control, either by removing the boundary layer by suction or else preventing boundary layer separation by blowing. This control, when applied to the wing tips, can be extremely beneficial in controlling tip stalls. A combination of a nose flap, split trailing-edge flap, and blowing air through a slot in the upper surface was the subject of a recent test, reference 39, on a 10% double wedge airfoil. The maximum value of the lift coefficient was

2.35, while for the basic airfoil the coefficient was 0.81.

Any attempt to summarize the effect and use of high lift devices on tailless aircraft must concern itself with the wing planform and airfoil shape. This is seen particularly in the use of trailing-edge split flaps where the effect of sweepback is of prime importance in determining the effectiveness of the flaps, since for sweep angles somewhat greater than 60° the advantages of the flap are completely lost, and the flaps may even become detrimental to airplane performance. The wing airfoil is important in that it is very difficult to incorporate mechanical devices in thin wings with very sharp leading and trailing edges. Since the specifications of high sweepback and thin wing apply directly to high speed aircraft it is seen that the design of an effective lift coefficient for these types can be extremely difficult, but nevertheless a necessity since the need for low drag at transonic and supersonic speeds leads to small wing area and high wing loading, necessitating high lift coefficients for landing.

Even the design of flaps for large airplanes with fairly low swee angles is not simple since a large flap deflected along the rear edge of wing would lead to a very strong pitching moment which the ailerons could not effectively or safely trim out at high lift coefficients and high angles of attack. This problem is also discussed in referenc 18 while one type of solution may be seen on the experimental airplane A.W.52, described in reference 40.

FUSELAGE AND FILLETS

The engineers who did the first experimental work on tailless aircraft had in their minds the idea of the complete flying wing, that is, the wing alone performing all the functions usually assigned to wing, fuselage, empennage, and nacelles. The realization of their dream is seen to vary quite a bit with the actuality of the present experimental tailless aircraft. While the very large wings have almost no fuselage or external drag producing elements, the small transonic and supersonic airplanes are rapidly approaching the shape of missiles, with a large body and relatively tiny supporting wing. Thus,the problems range from finding the shape of the most efficient wing to support a given internal lead, to finding an efficient wing to control the flight of a missile.

At the present time the question as to whether a fuselage shall be incorporated in the design of a tailless airplane depends primarily on space considerations. For example, if an airplane is contemplated to fly at a given gross weight and speed then the wing can be designed on this information, but the wing which will fit the specifications might not have enough free internal space to accommodate either the engine or other necessary items, such as a pilot, so provision must be made for a parasitic structure to supply the extra volume. The extra structure ranges from canopies, nacelles, or turrets, to entire fuselages. In general the large, slow speed airplanes can have nearly all their structure and equipment submerged into the wing, while the small ultra high speed craft must have a fuselage to carry engine and pilot. Because of the critical effect of the fuselage on the maximum speed of the latter airplane, it will be considered in more detail.

If the wing plan form can be determined even approximately for a supersonic airplane then the fuselage volume requirements can be determined. This is true because of the small percentage of the total volume contributed by a very thin and relatively small high speed wing. The fuselage shape must be such as to give least drag for the minimum wing-fuselage interference effect, and least detrimental effect on stability. In other words, the fuselage must be designed to give the maximum overall airplane performance, but not necessarily the least fuselage drag. From the standpoint of the fuselage alone, the design would be based on ballistic considerations, as given in reference 41, but the best fuselage shape to give minimum drag would include an extremely long nose cone which has an adverse effect on stability.

Some speculation, and work, has been done on the shape of the wing-fuselage intersection best suited for supersonic flight, and the conclusion drawn by the investigators was that the intersection should conform to the flow pattern of the air over the wing. The streamline shape for subsonic flight, calculated and tested in reference 42, seems to agree at least approximately to the shape useful for supersonic flight, qualitatively described in references 12 and 14, but in all cases the important fact is emphasized that the desired shape is a function of the lift coefficient and Mach number of the airplane in flight. Since the flight condition is not fixed the wing-fuselage junction could not work most efficiently at all times, but whether it could be designed to offset the flow disturbance at the junction of wing and fuselage for even a range of flight attitudes is not known. Furthermore the problem of combining the wing-fuselage intersection to a favorable fuselage design presents further problems, since the best condition for supersonic flow indicates a cylindrical body of revolution with as few surface contortions as possible. The cylinder with end cones, is the obvious choice since it gives the maximum cross sectional area for a given perimeter, and also is subject to drag computations. From reference 41 it seems that the best fuselage shape would have a round cylindrical body with a conical or ogival nose in front and a partial cone in back, if a jet exhaust is used. Formulas developed by von Karman indicate an ogival shape is better for a given length/diameter ratio for the nose, see references 41, 43, and 44. The wind tunnel tests of missiles are given in reference 45.

The design problem is further complicated by the necessity of having air ducts for turbo-jet or ram-jet engines. In large or slow speed airplanes the design of the duct is difficult but not critical, since the duct entrance is a small part of the total airplane geometry, but for high speed aircraft the opening and duct must be designed for efficient engine operation at low speeds. such as for landing, and also to give minimum drag and entrance loses at the design top speed of the airplane. The current manner is to place these ducts either at the nose of the airplane or ahead of the wing-fuselage juncture. The former method can reduce the overall fuselage length while still keeping a low effective cone angle, while the latter method has possibilities of preventing the formation of shock wave at the fillet.

RECOMMENDATIONS FOR FUTURE WORK

It is recommended that extensive tests be made to determine the characteristics of wings with sweepforward in order to form a comparison between swept-forward and swept-back wings. Also, tests should be made to find optimum airfoil shapes for supersonic flight, such airfoils in addition having good low speed characteristics.

Extensive work must be done to systematically evaluate high lift devices and control surfaces for highly swept wings of low aspect ratio.

In addition the effect of fuselage duct opening and wing-fuselage junction on drag coefficient must be known.

DESIGN OF SUPERSONIC TAILLESS AIRPLANE

To partially explain the use of the references given in the preceding sections the design of a tailless high-speed experimental airplane will now be considered. This type of airplane is used as an example because of the variety of problems to be overcome. These problems stem from the fact that the airplane must take off and land at low speeds, but also be able to fly as fast as possible. The latter requirement implies large sweepback, low aspect ratio, a small wing area, and a very thin airfoil section. The attempt must be made to reconcile these characteristics with the requirements for low speed lift and stability.

For simplicity the airplane will be considered to be an experimental prototype. It will carry no unnecessary equipment for flight except for 300 pounds of testing apparatus. Enough fuel will be carried to give an endurance of approximately one hour at full speed. The pilot must have some means for parachuting safely from the airplane at any speed, also an air conditioning system must be carried to protect the pilot from extremes of temperature. If possible the airplane will be fitted with a tricycle landing gear. The power will be supplied by a hypothetical turbo-jet engine, of 4000 pounds static thrust at sea level.

The preliminary estimate of the airplane specifications is made in the following manner. Since it is desired to have a very high top speed the airplane should be small and the weight low, with a high ratio of engine thrust to gross weight. If a ratio of 2/3 is taken, the weight would be about 6000 lbs. for an engine sea level static thrust of 4000 lbs. This ratio is considerably higher than that of conventional aircraft, but it is felt that it is justified in an experimental airplane subject to unknown hazards. A conventional value of wing loading should be used in view of the extremely poor maximum lift coefficients expected from the thin airfoils associated with supersonic flight. In other words, to attain even a landing speed of 130 m.p.h. it is necessary to use wing loadings of the order of 60 lbs/sq.ft. which is about the highest conventional value now used. Thus the combination of weight and wing loading lead to a wing with an area of 100 sq. ft.

As a basis for first estimates it was assumed that $C_D = 0.03$, a number based on past experience in the design of supersonic aircraft. If it is also assumed that at 50,000 feet altitude the engine thrust is 1500 lbs., then

$$q = \frac{T}{C_D \cdot S} = \frac{1500}{.03 \times 100} = 500 \ lbs/ft^2$$

This corresponds to a velocity of 1140 mph, or a Mach number of M = 1.72. To obtain a velocity component normal to the wing of M = 0.75 means that the wing must be swept back 65°. Since a taper ratio of the order of 1/2 will be used, for low speed characteristics, the leading edge of the wing can be considered to be swept back 65° without introducing very much error. After several attempts the wing shown in figure 4 was evolved. The wing has 98 square feet of area, a root chord of 10 feet, taper ratio of 0.513, aspect ratio of 1.70, and a span of 13 feet. The leading edge is swept back 65° while the 50% chord line is swept 61°.

It was felt that a delta wing would be less desirable than a tapered wing due to the very large percentage of wing area covered by the fuselage. The space covered on a delta wing would be even larger, and since the lift distribution across this combination of wing and fuselage is not yet known it was felt better to use the wing plan shown.

The diameter of the fuselage, forty inches, is the least that can fit around the engine, which is assumed to have a diameter of thirty-seven inches and to be fourteen feet long. The fuselage shape fore and aft was determined by both drag and stability considerations. The after cone has the best angle for minimum drag, 15°, reference 41, while the fore cone angle, also 15°, is a compromise between drag, stability, and visibility.

The entrance duct to the engine is located at the nose of the fuselage, the actual entrance diameter corresponding to the area required for an assumed flow of 73 lbs. of air per second at sea level at an airplane velocity of 300 M.P.H. This velocity was arbitrarily taken as a design criterion to give the best possible balance between low speed and high speed engine performance. The leading and trailing cone lengths were determined by their coincidence with the entrance and exit ducts, respectively.

The entrance duct must be divided to have the air flow around the pilot and then converge at the engine. The duct should be shaped so as to have as few friction and compression losses as possible. The most efficient duct for supersonic speed would have a converging section to slow the air down close to sonic velocities, then a diverging section to reduce it to the engine intake velocity.

The shape of the fillets, figure 4, was dictated more by a desire to incorporate a tricycle landing gear in the airplane than from the aerodynamic standpoint, though they should preserve the lift distribution over the fuselage and help prevent the formation of shock waves at the wing fuselage juncture. The fillets were made symmetrical, top and bottom, because of the small incidence of the wing.

The vertical tail was designed to have an area about one tenth the wing area, and leading edge angle equal to the wing's. This led to a tail of 10 sq. ft. in area, aspect ration of 0.75, and leading edge sweepback angle of 65°. The rudder chord was assumed to be 25% of the tail chord, though this is subject to experimentation.

Other factors that must be checked experimentally are the dimensions of the elevons and flaps. For a first approximation it was assumed the elevons extended along 30% of the semi-span, and the flaps were in the rest of the free space along the trailing edge. For both the chord was taken as 20% of the wing chord.

For ease of construction the airfoil was taken to be double wedge, of 10% thickness at the root and 4% at the tip, in each case measured to a chord perpendicular to the line joining the 50% chord points. This thickness was assumed, not for reasons of strength, but because it appeared to be about the minimum thickness that would allow the incorporation of the flaps and elevons, and the leading edge high lift device. The chord of the nose flap was taken as 10% of the wing chord.

To estimate the wing weight it is necessary to know the forces acting along the wing. If it is assumed that the lift distributions of figure 2 are representative, to a first approximation, of both subsonic and supersonic flight, then the spanwise forces can be calculated. For this case the curve of Cohen, and Weissinger, was assumed to represent the most likely condition. The results of the calculations appear in figure 3, for a unit weight, or lift of 1000 lbs. The critical condition will occur at the juncture of the wing and fillet when a total wing lift of 6000 lbs. is applied. A design factor of 18 is used to allow for unknown conditions. This factor is used to determine the design loads at the fillet. Knowing the loads, and thickness, the required skin area can be estimated, and thus the wing weight can be found to a first approximation. Calculations showed the thickness of the skin should be about 0.02" to 0.03" for steel, 0.07" to 0.08" for duraluminum, and 0.12" for magnesium. Since skin stiffness, or absence from wrinkles, is extremely important in supersonic flight, the use of the magnesium would be preferable, since the weight of the wing would in any case remain almost the same, namely, about 400 lbs.

The last point to determine is the position of the wing in relation to the fuselage. The a.c. of the wing can be approximated from the lift distribution curve of figure 2. Since the location of the a.c. is extremely doubtful for highly swept-back wings, it was assumed that the a.c. of any section was at the 25% point at subsonic speeds, and it was in relation to this point that the wing was located on the fuselage. It is realized that this is a procedure that is definitely arbitrary, but the justification lies in the fact that the center of gravity of the airplane must lie ahead of the center of pressure, which in no case will fall forward of the 25% point, for stable flight. It is very obvious that the above assumptions must be checked by means of wind tunnel tests at subsonic and supersonic velocities to determine the actual location and travel of the a.c.

The airplane weight and balance chart is shown in Table I, which indicates that for either loaded or unloaded flight the airplane is statically stable, since the fartherest forward position of the a.c. is approximated to be 151" from the nose.

The airplane drag estimate is shown in figure 5. This will be the minimum drag because it was assumed there would be no entrance duct drag or wing-fuselage interference effects.

The performance estimates of the airplane are shown on figure 6, for a sea level static thrust of 4000 lbs., while in figure 7 the estimate of performance is made for a sea level static thrust of 6000 lbs. It should be noted that the critical velocity near sea level will probably be limited by the increase in temperature of the airplane when flown at high Mach numbers.

At an altitude of 50,000 feet the endurance will be approximately fifty minutes at full speed, with the first engine in use.

TABLE I WEIGHT - BALANCE

Item	Weight - lbs.	Distance from nose - inches	Moment - inch l
Engine	2200	176	387,000
Pilot	200	60	12,000
Radio	30	22	660
Instruments	20	36	720
Test equipment	300	92	27,600
Fuselage	400	140	56,000
Rudder	60	294	17,640
Controls	30	140	4,200
Nose landing gear	70	90	6,300
Main landing gear	140	173.5	24,280
Air conditioner	80	92	7,360
Wing	400	116.5	46,600
Nose fuel	400	92	36,800
Tail fuel	420	260	109,200
Wing fuel	1000	130.5	130,500
Loaded condition	5750 lbs.	150.8 inches	866,860
Unloaded condition	3930 lbs.	150.4 inches	590,360

The wing location was arbitrarily selected so that the fartherest forward position of the center of pressure will be 151.0 inches from the nose of the airplane. Therefore,the airplane will be statically stable at all times.

PRELIMINARY WIND TUNNEL RESULTS

A 1/12-scale model of the airplane shown in figure 4 was tested in the Pasadena Junior College low speed wind tunnel. The model was the actual size of the drawing in figure 4. The intake duct and engine were simulated, approximately, by a round hole bored the length of the fuselage, the diameter of the hole being equal to the intake and exit duct diameter. Also, because of the very short length of time available for testing the model, the tests were made without fillets between wing and fuselage; therefore, the flaps, when used, extended inboard to the fuselage.

Four different tests were made. The original test was for the wing and fuselage, then wing and fuselage with a partial span 25% chord split flap, deflected 60°, extending inboard from the elevon position shown in figure 4. The third test was for the full span flap deflected 60°, where the elevon chord was also 25% of the wing chord. The last test was made with the full span split flap on the trailing edge of the wing and a 10% chord nose flap deflected 150°.

In each test the lift, drag, and pitching moment were measured. The dynamic pressure, q, of the tunnel was computed using static pressure readings and the tunnel calibration curve. The average tunnel speed was 106 ft/sec. The lift and drag coefficients were computed from

$$C_L = \frac{L}{qS} \quad , \quad C_D = \frac{D-d}{qS} \quad , \quad C_{M_x} = \frac{M-m}{qS\bar{c}}$$

where L is the lift measured, D is the drag measured, and M is the moment measured in the tests, while d is the drag due to the supporting struts, d = 0.01 q, and $\cancel{M}$ m is the tare moment.

The mean aerodynamic chord, $\bar{C}$, of the model was 0.653 feet and the model was supported at 0.651 $\bar{C}$. The location of the aerodynamic center was found by using the relation

$$C_{M_{a.c.}} = C_{M_x} - C_L\left(\frac{X}{C} - \frac{\delta}{C}\right)$$

If the $C_{M_{a.c.}}$ if constant, then

$$\frac{d(C_{M_x})}{d(C_L)} = \frac{X}{C} - \frac{\delta}{C}$$

where C_{M_x} is the moment coefficient referred to the supports and $\frac{X}{C}$ is the location of the wing support, while $\frac{\delta}{C}$ is the location of the a.c.

Plotting C_{M_x} against C_L gave $\frac{d(C_{M_x})}{d(C_L)} = 0.308$ for zero C_L, therefore $\frac{\delta}{C} = 0.343$. Since in the original design it was assumed that the a.c. would be at $\frac{\delta}{C} = 0.250$, it is apparent that the wing is located too far behind the center of gravity. Hence, though the airplane is stable, the moments required to trim it for level flight would be unnecessarily high.

The moments about the a.c. were calculated from

$$C_{M_{a.c.}} = C_{M_x} - 0.308\, C_L$$

Knowing the pitching moment coefficient, the complete wind test results are plotted in figure 8.

Comparison of the results to preliminary estimates indicate several points that affect the design of the airplane. The most important point is that the airplane is unstable at relatively high lift coefficients; therefore, modifications must be made in the design to eliminate this instability. The other factor is that the small slope of the lift curve necessitates either a very high angle of attack for reasonable landing speeds or else landing speeds somewhat higher than ordinary.

Another interesting feature is that higher lift coefficients can be obtained with trailing edge split flaps, though previous references had indicated that this would not be the case. The continued increase of the lift coefficient with angle of attack is interesting in respect to the tuft studies, figure 9. Though tufts were used on the wing for all configurations the results seemed to be about the same, so only one set is shown. The flow over the wing was uniform and parallel to the free stream flow at negative and very small positive angles of attack, wing A. At some angle between 2^{o} and 5^{o} the air would suddenly flow along the outer half of the leading edge of the wing, and remain nearly parallel elsewhere. The turbulent action of the tufts indicated that that part of the wing was stalled in the spanwise direction, wing B. For all angles of attack greater than the critical angle the tufts gradually assumed the characteristics of wing C, that is, more and more of the outer portion of the wing in the stalled condition. At no angle of attack up to 34^{o} did the trailing edge of the wing appear to be stalled.

REFERENCES

1. Lee, G. H.: The Case For the Tailless Aircraft. Journal of the Royal Aeronautical Sciences, Nov. 1946.

2. Soulé, Hartley A.: Influence of Large Amounts of Wing Sweep on Stability and Control Problems of Aircraft. N.A.C.A. T.N. 1088, 1946.

3. Seacord, Charles L. Jr. and Ankenbruck, Herman O.: Determination of the Stability and Control Characteristics of a Straight-Wing, Tailless Fighter Airplane Model in the Langley Free-Flight Tunnel. N.A.C.A. ARR No. L5K05, 1946.

4. Seacord, Charles L. Jr. and Ankenbruck, Herman O.: Effect of Wing Modifications on the Longitudinal Stability of a Tailless All-Wing Airplane Model. N.A.C.A. ARR No. L5G23, 1945.

5. Shortal, Joseph A. and Maggin, Bernard: Effect of Sweepback and Aspect Ratio on Longitudinal Stability Characteristics of Wings at Low Speeds. N.A.C.A T.N. 1093, 1946.

6. Letko, William and Goodman, Alex.: Preliminary Wind-Tunnel Investigation at Low Speed of Stability and Control Characteristics of Swept-Back Wings. N.A.C.A. T.N.1046, 1946.

7. Göthert, B.: High Speed Measurements on a Swept-Back Wing (Sweepback Angle = 35°). Translation in N.A.C.A. T.M. 1102, 1947.

8. Jones, Robert T.: Wing Plan Forms for High Speed Flight. N.A.C.A. T.N. 1032, 1946.

9. Jones, Robert T.: Thin Oblique Airfoils at Supersonic Speed. N.A.C.A. T.N. 1107.

10. Lindsey, W. F., Daley, Bernard N., and Humphreys, Milton D.: The Flow and Force Characteristics of Supersonic Airfoils at High Subsonic Speeds. N.A.C.A. T.N. 1211, 1947.

11. Hilton, W. F. and Pruden, F. W.,: Subsonic and Supersonic High Speed Tunnel Tests of Faired Double Wedge Airfoil. British R & M. No. 2057, 1943.

12. Ackeret, J.: Airforces on Airfoils Moving Faster than Sound. N.A.C.A. T.M. 317, 1925.

13. Ivey, Reese H.: Notes on the Theoretical Characteristics of Two-Dimensional Supersonic Airfoils. N.A.C.A. T.N. 1179, 1947.

14. Bonney, E. Arthur: Characteristics of Rectangular Wings at Supersonic Speeds. Journal Aeronautical Sciences, Feb., 19

15. Puckett, Allen E.: Supersonic Wave Drag of Thin Airfoils. Journal Aeronautical Sciences, Sept. 1946.

16. Lagerstrom and Wall: Formulas in Three-Dimensional Wing Theory (1). Douglas Aircraft Company S.M. Report 11901, July, 194

17. Hayes, W. D., Browne, S. H., and Lew, R. J.: Linearized Theory of Conical Supersonic Flow with Application to Triangular Wings. North American Aviation Report No. NA-46-818, Sept. 1946.

18. Snow, R. M. and Bonney, E. A.: Aerodynamic Characteristics of Wings at Supersonic Speeds. Bumble Series Report No. 55, Johns Hopkins University Applied Physics Laboratory, March, 1947.

19. Stability Research Division: An Interim Report on the Stability and Control of Tailless Airplanes. NACA ACR L4H19, 1944.

20. Thiel, A. and Weissinger, J.: Pressure Distribution Measurements on a Straight and on a 35° Swept-Back Tapered Wing. N.A.C.A T.M. 1126, 1947.

21. Pearson, Henry A. and Jones, Robert T.: Theoretical Stability and Control Characteristics of Wings with Various Amounts of Taper and Twist. N.A.C.A. T.R. 635, 1938.

22. Campbell, John P. and Seacord, Charles L., Jr.: Determination of the Stability and Control Characteristics of a Tailless All-Wing Airplane Model with Sweepback in the Langley Free-Flight Tunnel. N.A.C.A. ARR L5A13, 1945.

23. Anderson, Ramond F.: A Comparison of Several Tapered Wings Designed to Avoid Tip Stalling. N.A.C.A. T.N. 713, 1939.

24. Jones, Robert T.: Properties of Low-Aspect-Ratio Pointed Wings at Speeds Below and Above the Speed of Sound. N.A.C.A. T.N. 1032, 1946.

25. Jones, Robert T.: Notes on the Stability and Control of Tailless Aircraft. N.A.C.A. T.N. 837, 1941.

26. Pitkin, Marvin and Maggin, Bernard: Analysis of Factors Affecting Net Lift Increment Attainable with Trailing-Edge Split Flaps on Tailless Airplanes. N.A.C.A. ARR No. L4I18, 1944.

27. Anderson, Raymond F.: Determination of the Characteristics of Tapered Wings. N.A.C.A. T.R. 572, 1936.

28. Garbell, M. A.: Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings. Journal Aeronautical Sciences, Oct. 1946.

29. Pearson, Henry A. and Anderson, Raymond F.: Calculation of the Aerodynamic Characteristics of Tapered Wings with Partial Span Flaps. N.A.C.A. T.R. 665, 1939.

30. Maggin, Bernard and Shanks, Robert E.: The Effect of Geometric Dihedral on the Aerodynamic Characteristics of a 40° Swept-Back Wing of Aspect Ratio 3. N.A.C.A. T.N. 1169, 1946.

31. Weissinger, J.: The Lift Distribution of Swept-Back Wings. Translation in N.A.C.A. T.M. 1120, 1947.

32. Cohen, Doris: Theoretical Distribution of Load over a Swept-Back Wing. N.A.C.A. 1942.

33. Mutterperl, W.: The Calculation of Span Load Distribution on Swept-Back Wings. N.A.C.A. T.N. 834, 1941.

34. Garbell, Maurice A.: Theoretical Principals of Wing-Tip Fins for Tailless Airplanes and Their Practical Applications. Journal Aeronautical Sciences. Oct. 1946.

35. Bennett, Charles V. and Johnson, Joseph L.: Experimental Determination of the Damping in Roll and Aileron Rolling Effectiveness of Three Wings Having 2°, 42°, and 62° Sweepback. N.A.C.A. T.N. 1278, 1947.

36. Harper, Charles W. and Jones, Arthur L.: A Comparison of the Lateral Motion Calculated for Tailless and Conventional Airplanes. N.A.C.A. T.N. 1154, 1947.

37. Lemme, H. G.: Force and Pressure Distribution Measurements on a Rectangular Wing with a Slotted Droop Nose and with either Plain and Split Flaps in Combination with a Slotted Flap. Translation in N.A.C.A. T.M. 1108, 1947.

38. Krueger, W.: Systematic Wind Tunnel Measurements on a Laminar Wing with Nose Flap. Translation in N.A.C.A. T.M. 1119, 1947.

39. Pollack, A. D. and Reck, F. F.: A Study of Methods to Increase the Lift of Supersonic Airfoils at Low Speeds. California Institute of Technology Professional Thesis, 1947.

40. Twin-Jet A.W. 52: Flight and Aircraft Engineer, Dec. 19, 1946.

41. Charters, A. C.: Some Ballistic Contributions to Aerodynamics. Journal Aeronautical Sciences, March, 1947.

42. Watkins, Charles E.: The Streamline Pattern in the Vicinity of an Oblique Airfoil. N.A.C.A. T.N. 1231, 1947.

43. Von Karman, Th.: The Problem of Resistance in Compressible Fluids. Reale Accademia D'Italia, Roma, 1936.

44. Von Karman, Th. and Moore, Norton B.: Resistance of Slender Bodies Moving with Supersonic Velocities, with Special Reference to Projectiles. Transactions of A.S.M.E., 1932.

45. Walchner, O.: Systematic Wind-Tunnel Measurements on Missiles. Translation in N.A.C.A. T.M. 1133, 1947.

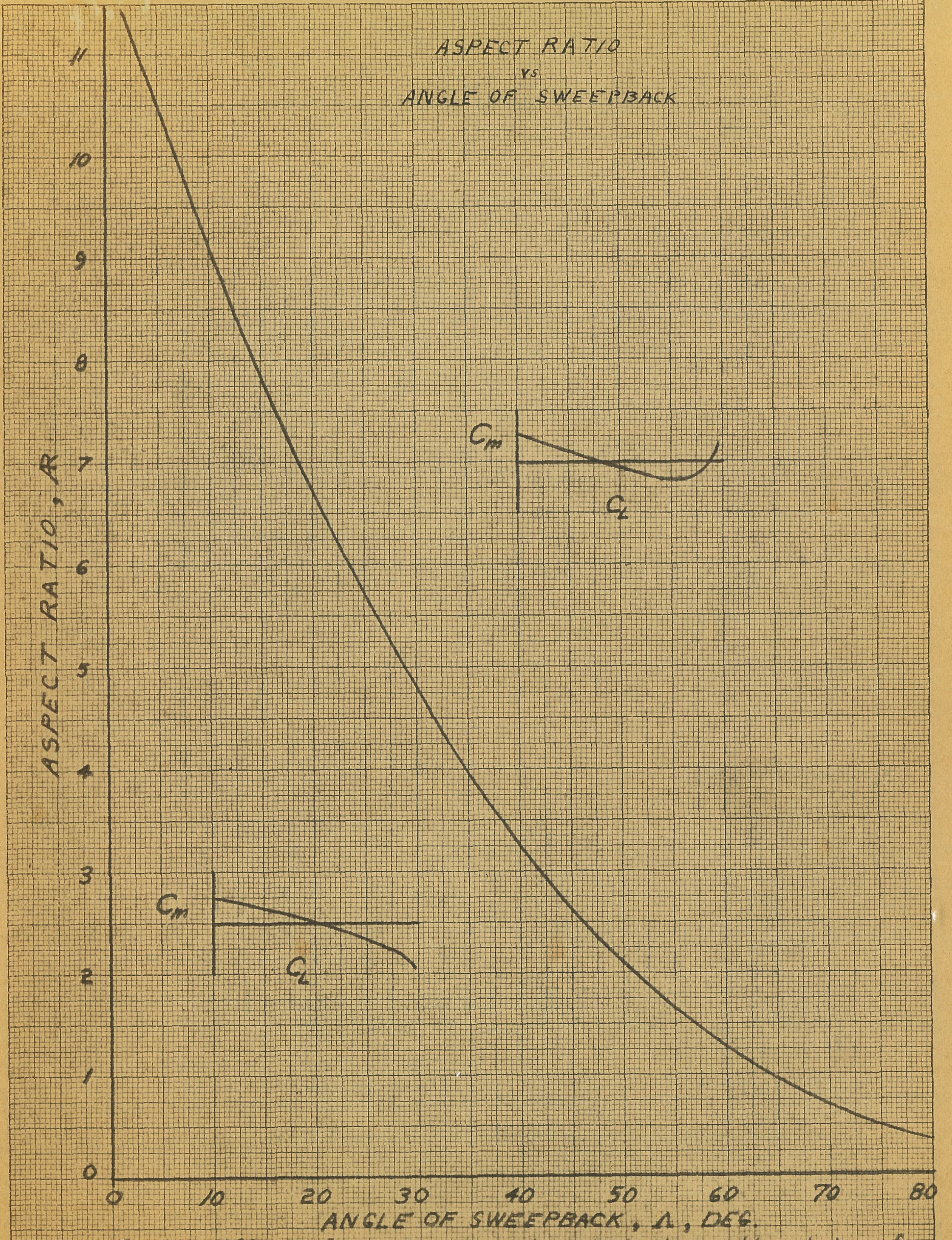
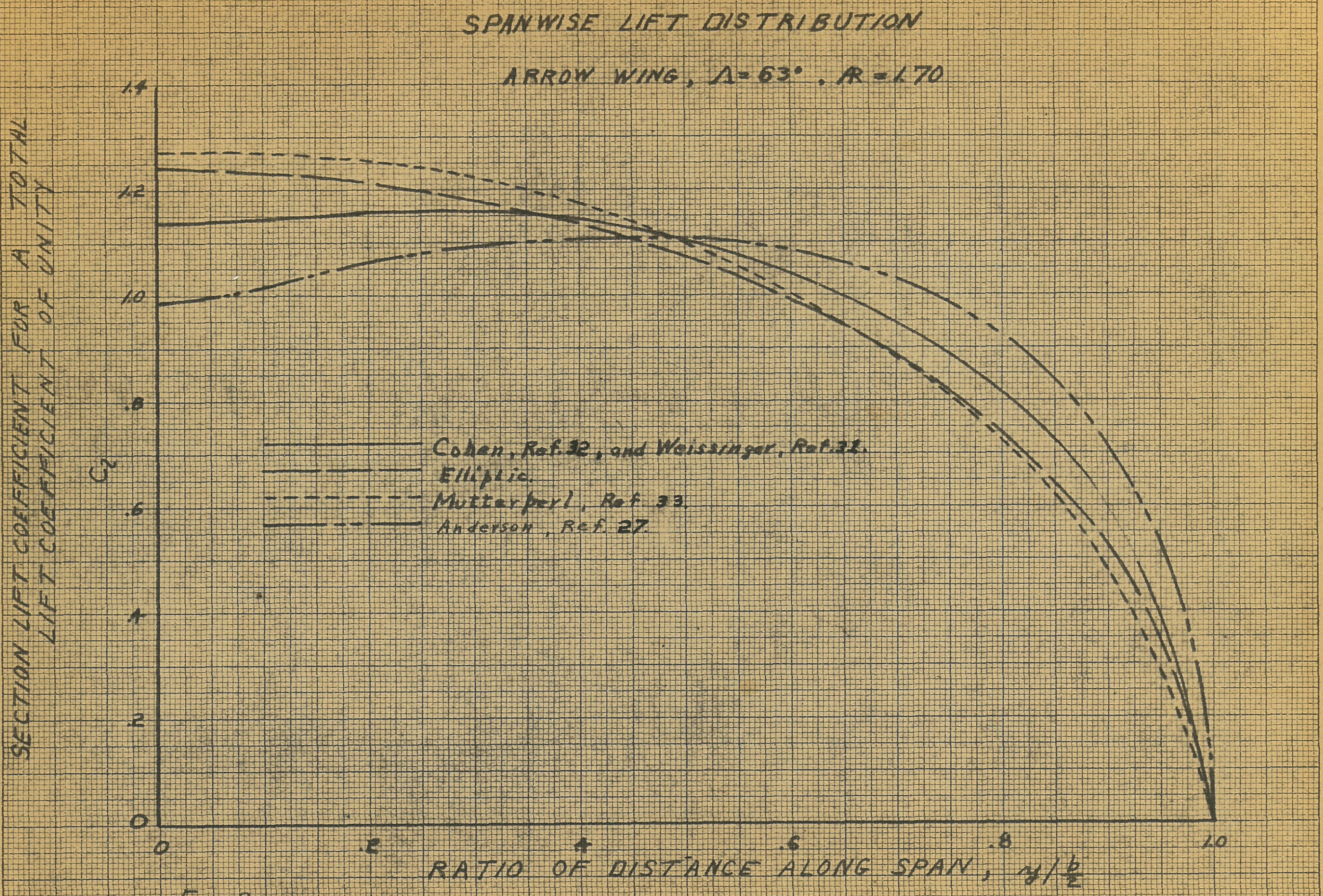


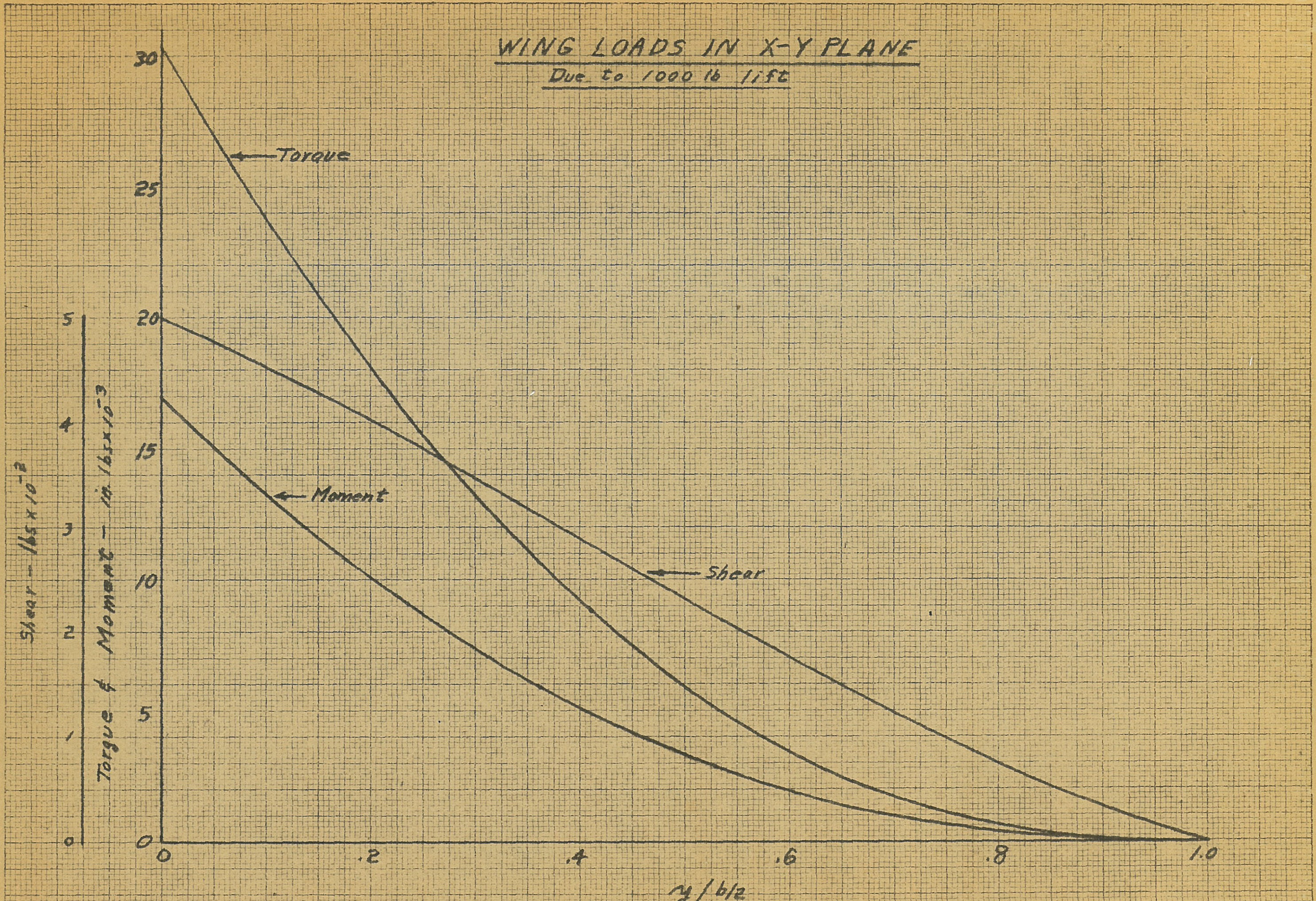
Fig. 1.– Effect of aspect ratio and sweepback on the shape of the pitching-moment-coefficient curve at the stall.

References 4, 5, 6.



SPANWISE LIFT DISTRIBUTION
ARROW WING, Λ = 63°, Æ = 1.70
SECTION LIFT COEFFICIENT FOR A TOTAL LIFT COEFFICIENT OF UNITY
C_l
Cohen, Ref. 32, and Weissinger, Ref. 31.
Elliptic
Mutterperl, Ref. 33
Anderson, Ref. 27
RATIO OF DISTANCE ALONG SPAN, $y/\frac{b}{2}$
0
.2
.4
.6
.8
1.0
1.2
1.4

Fig-2.



WING LOADS IN X-Y PLANE
Due to 1000 lb lift
Torque
Moment
Shear
30
25
20
15
10
5
0
5
4
3
2
1
0
Torque & Moment - in lbs x 10^-3
Shear - lbs x 10^-2
0
.2
.4
.6
.8
1.0
y / b/2

Fig.-3.

Dore_F_1947
Fig. 4

SUPERSONIC
TAILLESS AIRPLANE

Scale: 1" = 1'

[Full scale for wind tunnel model]

Fig. 4.

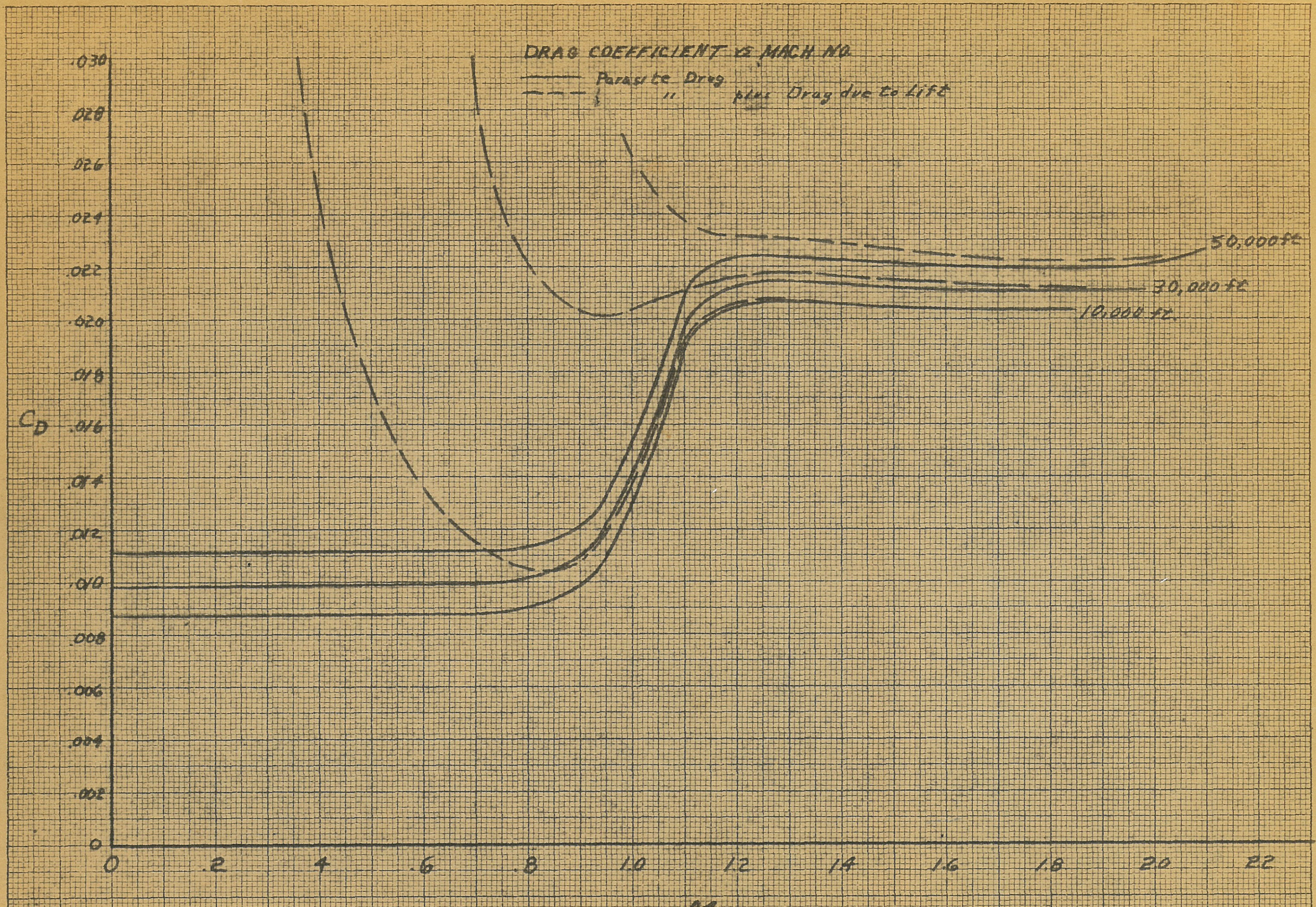

DRAG COEFFICIENT vs MACH NO.
Parasite Drag
" plus Drag due to Lift
50,000 ft
30,000 ft
10,000 ft.
C_D
.030
.028
.026
.024
.022
.020
.018
.016
.014
.012
.010
.008
.006
.004
.002
0
.2
.4
.6
.8
1.0
1.2
1.4
1.6
1.8
2.0
2.2
M

Fig-5

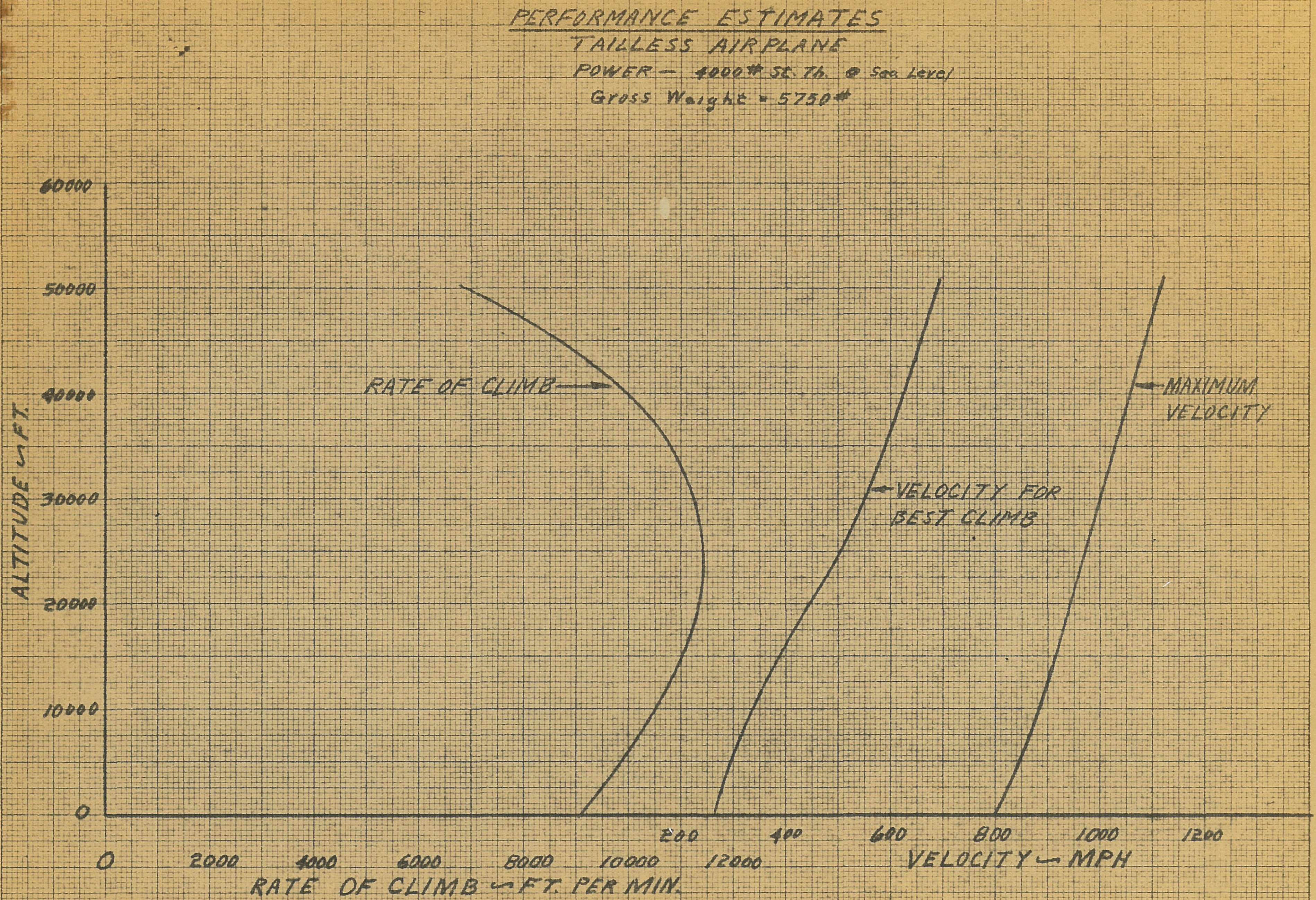

PERFORMANCE ESTIMATES
TAILLESS AIRPLANE
POWER — 4000# St. Th. @ Sea Level
Gross Weight = 5750#
RATE OF CLIMB
VELOCITY FOR BEST CLIMB
MAXIMUM VELOCITY
ALTITUDE — FT.
60000
50000
40000
30000
20000
10000
0
0
2000
4000
6000
8000
10000
12000
RATE OF CLIMB — FT. PER MIN.
200
400
600
800
1000
1200
VELOCITY — MPH

Fig. 6.

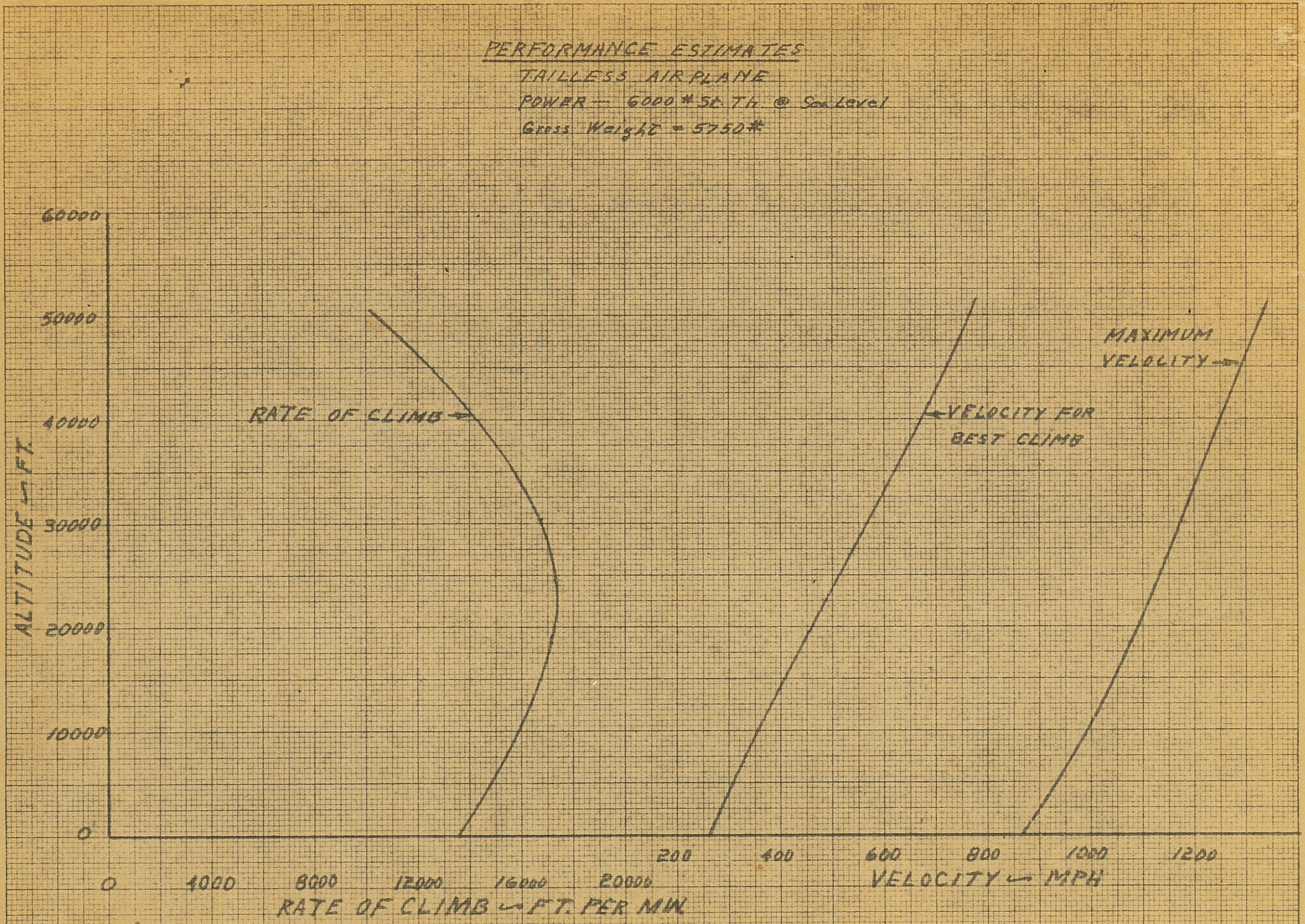

PERFORMANCE ESTIMATES
TAILLESS AIRPLANE
POWER — 6000# St. Th. @ Sea Level
Gross Weight = 5750#
ALTITUDE — FT.
60000
50000
40000
30000
20000
10000
0
RATE OF CLIMB
VELOCITY FOR BEST CLIMB
MAXIMUM VELOCITY
0
4000
8000
12000
16000
20000
RATE OF CLIMB — FT. PER MIN.
200
400
600
800
1000
1200
VELOCITY — MPH

Fig. -7

TUFT STUDIES

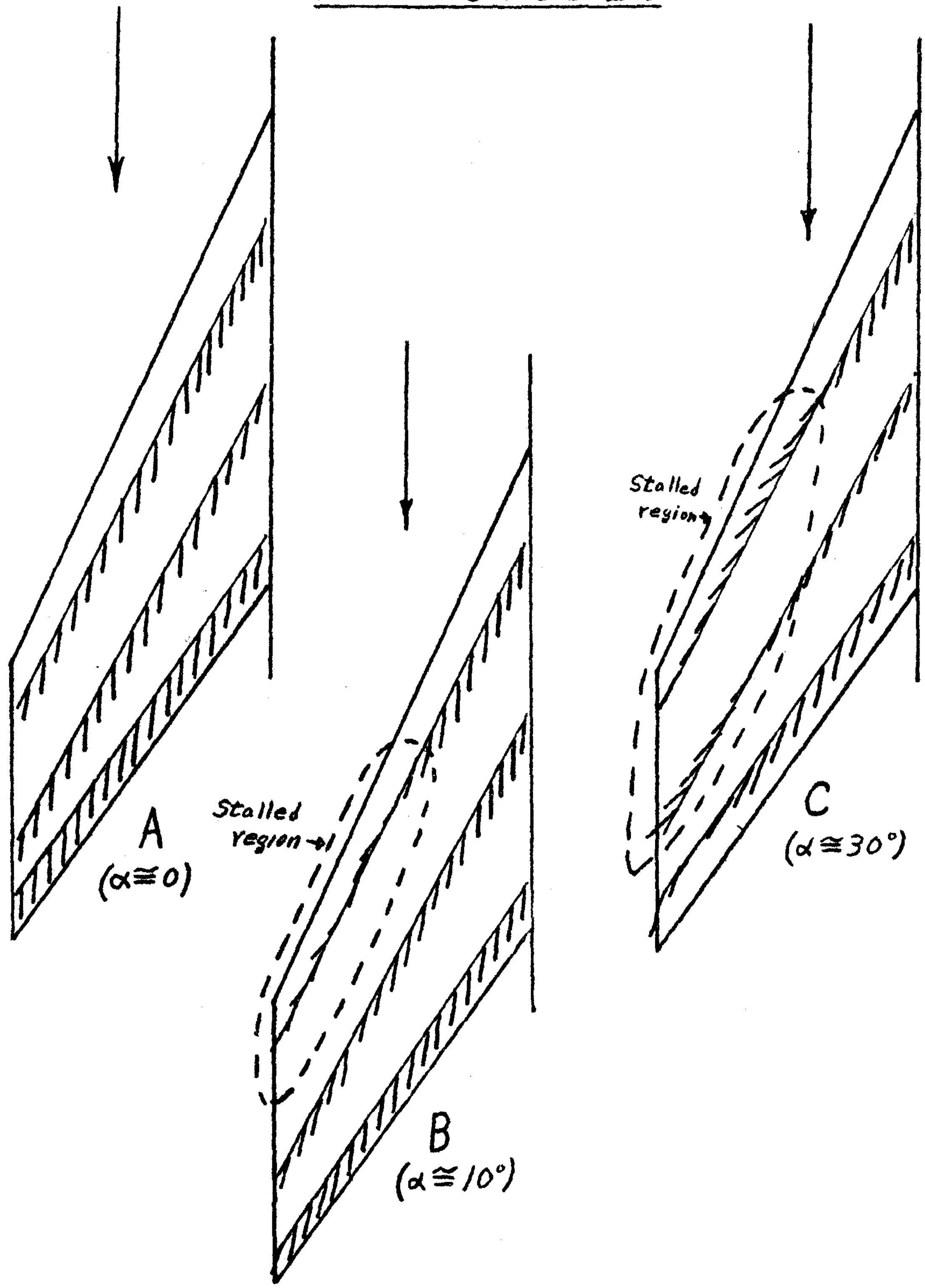
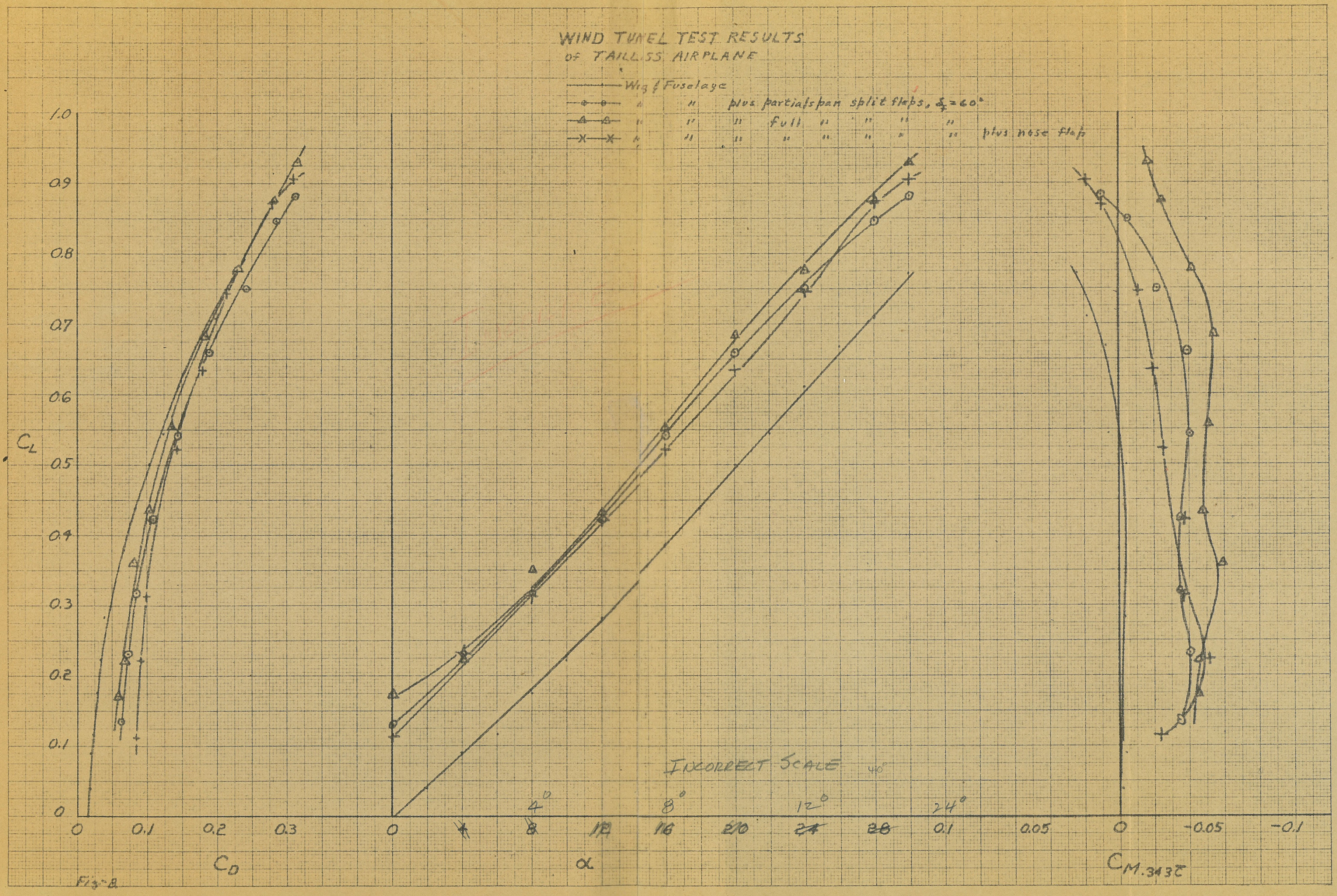


Fig. 9.


WIND TUNEL TEST RESULTS
OF TAILLSS AIRPLANE
Wig & Fuselage
" " plus partialspan split flaps, δ_f=60°
" " " full " " " "
" " " " " " " " plus nose flap
C_L
C_D
α
INCORRECT SCALE
$C_{M.343\bar{c}}$
Fig-8