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VORTEX LIFT RESEARCH: EARLY CONTRIBUTIONS AND SOME CURRENT CHALLENGES

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SUMMARY

This paper briefly reviews the trend towards slender-wing aircraft for supersonic cruise and the early chronology of research directed towards their vortex-lift characteristics. An overview of the development of vortex-lift theoretical methods is presented, and some current computational and experimental challenges related to the viscous flow aspects of this vortex flow are discussed.

INTRODUCTION

Beginning with the first successful controlled flights of powered aircraft, there has been a continuing quest for ever-increasing speed, with supersonic flight emerging as one of the early goals. The advantage of jet propulsion was recognized early, and by the late 1930's jet engines were in operation in several countries. High-speed wing design lagged somewhat behind, but by the mid 1940's it was generally accepted that supersonic flight could best be accomplished by the now well-known highly swept wing, often referred to as a "slender" wing. It was also found that these wings tended to exhibit a new type of flow in which a highly stable vortex was formed along the leading edge, producing large increases in lift referred to as vortex lift. As this vortex flow phenomenon became better understood, it was added to the designers' options and is the subject of this conference.

The purpose of this overview paper is to briefly summarize the early chronology of the development of slender-wing aerodynamic technology, with emphasis on vortex lift research at Langley, and to discuss some current computational and experimental challenges.

TOWARDS SLENDER-WING AIRCRAFT

I joined the Langley staff in July of 1944, shortly after Allied pilots had first encountered the German swept wing Me 262 jet fighter shown in figure 1. Since prototype jet aircraft had been built and flown previously by the Germans, British, and Americans, the most surprising feature of the Me 262 was its sweptback wing which contributed to a speed advantage through a delay of the onset of compressibility drag - a benefit of sweep not understood in the Allied nations at the time. Although the 18° of sweepback was the fortuitous result of a design change, in 1940, to fix a center-of-gravity problem, German researchers had, that same year, demonstrated in the wind tunnel that Busemann's 1935 supersonic swept wing theory (ref. 1) also applied to subsonic compressibility effects (ref. 2) and immediately began the design of more highly swept wings for the Me 262. Thus, the Me 262 program represents the genesis of the trend towards slender-wing supersonic aircraft as illustrated in figure 2. The Me 262 had first flown in 1942, and advanced versions incorporating wings with sweep angles as high as 50° were studied (ref. 3). A 40° sweep version, shown in figure 2, had been tested in a German wind tunnel in 1941 and reached the prototype stage in early 1945 but was accidentally destroyed on the runway before its first flight (ref. 4).

Highly swept delta wings were also being studied in Germany but none reached the powered prototype stage, and the conclusion of the war brought an end to the studies.

The benefit of sweep with regard to high-speed flight remained a mystery outside of Germany until January 1945 when R. T. Jones of the Langley Research Center completed a theoretical study in which he demonstrated, independent of Busemann's work, that wing pressure distributions are determined solely by the "component of motion in a direction normal to the leading edge." He further pointed out that for efficient supersonic flight, the wing should be swept behind the Mach cone with the sweep angle being such that the normal component of velocity is below the airfoil's critical speed (ref. 5).

Jones' theoretical work and the subsequent acquisition of German swept wing data stimulated extensive swept wing research programs at Langley and Ames and aircraft development programs within the Air Force and industry. Two of the early U. S. aircraft utilizing the concept are shown on the right of figure 2. In 1947 the North American XP-86, which utilized some of the 40° swept wing data from the Me 262 program, made its first flight and became the first of a long line of swept wing jets optimized for high subsonic cruise and capable of supersonic dash (ref. 6). The Convair XF-92A, which first flew in 1948, represents the beginning of the evolution of the slender wings desirable for efficient supersonic cruise of interest to this paper (ref. 7).

While the slender wing combined with the jet engine made supersonic flight practical, its high-speed benefits did not come without a sacrifice in subsonic capabilities. It was, of course, recognized early that the slender, low-aspect ratio planforms required for efficient supersonic flight provided extremely poor subsonic performance due to their high level of induced drag. Their lift gradient was low, and it was found that the effectiveness of conventional high-lift flow control devices was poor. It was obvious that new design approaches were needed.

For those aircraft missions requiring very high levels of both subsonic and supersonic performance, the most obvious solution was the application of adjustable planform geometry in the form of variable sweep. The first wind tunnel study of symmetrical variable sweep appears to be that carried out in Langley's 300-MPH, 7- by 10-ft High-Speed Tunnel beginning in 1946. Further analytical research and experiments in the 7- by 10-ft High-Speed Tunnel in 1958 provided the variable sweep concept that led to the F-111, F-14, and B-1 aircraft (see refs. 8 and 9 for reviews).

THE DISCOVERY OF VORTEX LIFT

Another important event related to the application of slender-wing benefits to supersonic aircraft also took place in 1946 when researchers at Langley discovered a flow phenomenon that was to play an important role in the design of fixed planform slender-wing aircraft not requiring the high degree of multimission capability offered by variable sweep. This phenomenon was the leading-edge vortex flow which is the subject of this conference.

The sequence of events leading to this discovery began in 1945 when American aerodynamists surveying German aeronautical developments decided that the Lippisch highly swept delta wing DM-1 test glider, shown on the left in figure 3, should be shipped to Langley for tests in the Full-Scale Tunnel. The Germans had planned to use the DM-1 for flight studies of the low-speed characteristics of a proposed supersonic aircraft (refs. 10 and 11). While the American team recognized that the wing was too thick for efficient supersonic flight, they felt that it offered an early opportunity to study the low-speed characteristics of highly swept delta wings under full-scale conditions and arrangements were initiated by a letter dated November 17, 1945 (United States Air Force in Europe to Commanding General, Army Air Forces, Washington, D.C., 1945).

The glider arrived at Langley early in 1946 and is shown in the Full-Scale Tunnel on the right of figure 3. It will be noted that several changes had been made, and the sharp leading edge shown in the photograph was the result of Langley research to improve the high lift characteristics. The Langley study, reported by Wilson and Lovell (ref. 12), discovered that the maximum lift of the original round leading-edge configuration was considerably lower than that obtained on similar wings previously tested at low Reynolds numbers in Germany and at Langley. A small model was quickly built and its flow characteristics studied. It was found that at low Reynolds numbers, laminar separation occurred at the leading edge, and a strong vortex developed which produced large lift increments. It was then reasoned that a sharp leading edge would produce a similar flow even at high Reynolds numbers, and the DM-1 was modified as shown. The results shown in figure 4 produced large vortex lift increments which offered a solution to the slender-wing high angle-of-attack lift capability problem. Their research provided the first insight on the effects of leading-edge radius and Reynolds number on vortex lift. A "cross-flow separation" model of the vortex flow was also proposed in their paper. Although this research remained under a security classification for four years, it did provide a stimulus for additional research at Langley and Ames and interest within the Air Force and the aircraft industry. Much of the international interest in vortex flow was generated somewhat later through its independent discovery by French researchers during studies carried out in 1951 and 1952, closely followed by related research in Great Britain (refs. 13 and 14).

THE CONTROLLED SEPARATION CONCEPT

Two of the primary characteristics of slender wings are illustrated in figure 5. One of the primary driving forces in the designers' selection of a slender wing for supersonic cruise is the reduction in lift-dependent drag shown on the left for a Mach number of 2.5. As pointed out by both Busemann (ref. 1) and Jones (ref. 5), sweeping the leading edge behind the Mach cone provides a subsonic type flow with upwash manifesting itself as a leading-edge thrust effect, as long as the flow remains attached, which more than offsets the adverse effect of aspect ratio reduction. However, as shown on the right, attached flow theory predicted the subsonic lift capability to be very low for a slender wing. The wing shown has a sweep of 75^{o} selected for a cruise point of about M=2.5. If, however, the flow separates at the leading edge, vortex flow develops, and large vortex-lift increments are attained. This lift, associated with the large mass of air accelerated downward by the nonplanar vortex sheets, greatly relieves the lift deficiency of slender wings with attached flow. With a sharp leading edge, the separation occurs simultaneously along the edge and, thereby, eliminates the spanwise stall progression which produces various stability and control problems. In addition, vortex-induced reattachment delays trailing-edge separation. Competing with these advantages, of course, is the increased drag resulting from the loss of leading-edge thrust.

The above slender-wing characteristics led to a new aircraft design concept which departed from the time honored "attached flow" wing design for certain flight conditions. Basically, this concept consisted of designing the wing for attached flow at supersonic cruise conditions using concepts such as conical camber with pressure components providing the leading-edge thrust effect but allowing the flow to separate at the leading edge and generate vortex lift to provide the low-speed lift required. This simplified the wing design by reducing the need for leading- and trailing-edge high-lift flow-control devices—devices which are relatively ineffective on highly swept wings and increase the complexity and weight.

The U. S. supersonic delta-wing aircraft designed in the mid 1950's to the early 1960's utilized this approach to various degrees as illustrated in figure 6 with a photograph of the B-70 in the landing mode. The photograph, taken around 1965, illustrates the strong leading-edge vortices generated by the thin, 65.5^{o} delta wing and made visible by natural condensation. However, during this period, NASA's basic and applied research on leading-edge vortex flows lagged considerably behind that of Great Britain and France where researchers were enthusiastic over what many of them described as the "new aerodynamics." In 1962, an agreement was signed between the British and French to develop a supersonic commercial transport, and they soon agreed on the now well-known slender ogee-delta wing planform with the design based on the "controlled flow separation" concept. They improved the application of the design concept by detailed tailoring of the wing warp and planform to improve the performance of both the attached flow and vortex flow modes as well as the transition mode.

The result of this extensive development program was the remarkable "Concorde" supersonic commercial transport which is still the only supersonic transport in regular passenger service, although the Soviets are undoubtedly amassing considerable experience in their TU-144 flight programs. For details of the "Concorde" development, the reader is referred to references 14 and 15.

RENEWED LANGLEY INTEREST

Langley research related to vortex lift began to accelerate in the mid 1960's. Contributing to this acceleration was the interest generated by the extensive research in France and Great Britain in support of the Concorde, growing interest in supersonic cruise aircraft and lightweight highly maneuverable fighter aircraft, and the development at Langley of a three-dimensional theoretical approach that provided an improved understanding of leading-edge vortex flows for arbitrary planforms.

Vortex-Lift Theory Development

The three-dimensional theory referred to above is the leading-edge suction analogy developed at Langley in 1966 (ref. 16). Prior to this development, the theoretical approaches were generally confined to slender-wing conical flow approximations in order to simplify the nonlinear system of equations resulting from the fact that neither the strength or shape of the free vortex sheet is known.

A chronology of some of the advances that have been made in the development of theoretical methods for predicting the aerodynamics of sharp-edged slender wings having leading-edge vortex flow is presented in figure 7. The total lift developed on a 76^{0} delta wing as a function of angle of attack is used to illustrate the advances that have been made, and both experimental measurements and attached flow calculations are included for comparison purposes. Shown are three of the conical flow theories and two of the nonconical, or three-dimensional, theories.

The first mathematical model of the vortex flow was proposed and investigated by Legendre at ONERA in France in 1952 (ref. 17). Using a slender-body approach, he represented the leading-edge vortex sheets by two isolated vortices and solved for their position and strength by applying a Kutta condition at the leading edge and requiring that the vortices sustain no force. While this approach did produce a nonlinear vortex lift, the simplifying assumptions resulted in a greatly overpredicted lift force.

Improvements in Legendre's approach followed, and in 1955 Brown and Michael of the Langley Research Center replaced the no-force condition on the vortex by one on the vortex and a feeding sheet, taken together, which provided some improvement in the vortex lift prediction (ref. 18).

By the mid 1960's, many conical flow theories had been developed drawing on slender-wing concepts, the most notable of which was that of Smith of the RAE in England which, although still overpredicting the lift, provided an excellent representation of the spiral-shaped vortex sheet (ref. 19). These theories provided much insight into the vortex flow phenomena and contributed to early design concepts. However, their applicability to wings of practical interest was limited by their exclusion of effects such as, for example, the trailing-edge Kutta condition at subsonic speeds and the proximity of the Mach cone at supersonic speeds.

The difficulties in accounting for these three-dimensional effects were greatly relieved by the development of the "leading-edge suction analogy" in 1966 at the Langley Research Center (ref. 16). This analogy equates the normal force produced by the separation induced vortex flow to the attached flow leading-edge suction force. This allows three-dimensional linearized flow theory to be used for this nonlinear flow phenomenon thereby greatly reducing theoretical complexity as well as numerical run time and cost. An indication of its ability to overcome the limitations of slender-wing conical flow theories is illustrated in figure 7 for the low-speed case. Excellent agreement was also obtained for both lift and drag for a wide range of delta wings up to angles of attack where vortex breakdown or vortex asymmetry occur.

It was soon found that the suction analogy offered a broad range of prediction capability and the possibility of design-by-analysis capability. The method, therefore, was used to develop a coordinated theoretical-experimental vortex flow research program by a small group of Langley researchers, which will be described in the following section. However, before leaving the theory chronology, it should be pointed out that the researchers recognized the eventual need to provide a method that models the complete flow field and establishes surface pressure details. Therefore, they contracted with the Boeing Company in 1973 to develop a higher order panel method to model the leading-edge vortex flow. A schematic of the resulting theoretical model, known as the "free vortex sheet" (FVS) method, is shown on the right of figure 7 and will be described later.

Also supported was the development of some free vortex filament approaches in the university community to determine if they might provide a simpler method that would satisfy the design and analysis needs. However, the experience gained from these and other studies indicated that the filament formulations have failed to provide consistent and accurate load distributions, they exhibit undesirable numerical modeling sensitivity, and they are unduly complicated for the estimation of overall force/moment properties. A similar conclusion has been drawn by Hoeijmakers in reference 20, and no further reference to these methods will be made in this paper.

The Langley Research Program

The suction analogy provided a vortex flow analysis tool that included three-dimensional effects and offered the designer the possibility of at least some limited design capability. With this new theoretical tool and their renewed vortex flow interest, the Langley researchers initiated a coordinated theoretical and experimental vortex flow research program. Although the bulk of their experimental research was performed at subsonic speeds in the 7- by 10-foot high-speed tunnel, they extended their studies to transonic, supersonic, and hypersonic speeds by scheduling time in other Langley facilities.

The purpose of this section is to briefly review the early years of this program which covered both performance-and stability-related vortex flow characteristics. For a more complete review of the program, the reader is referred to the summary papers by Lamar and Luckring (ref. 21) and by Lamar and Campbell (ref. 22).

Performance Characteristics - The initial application of the suction analogy illustrated in figure 7 was for incompressible flow, and it was found to provide excellent predictions of the lift and drag of sharp-edged delta wings over a large range of sweep angles and angles of attack (see refs. 16 and 23).

Since the analogy could be applied using the attached flow leading-edge suction from any accurate, attached flow theory, it was extended into the high subsonic and supersonic ranges early in the program (ref. 24). An example of this application is shown in figure 8 for a 76° delta wing at an angle of attack of 18°. The analogy was applied in the subsonic range using the Prandtl-Glauert transformation and in the supersonic range using linearized supersonic theory. The resulting vortex lift increments C_{L_V} are shown on the left compared with experiment, and agreement is excellent over the entire Mach range. The experimental values were obtained by subtracting attached flow theory values from the total measured lift. The results illustrate the ability of the analogy to predict the reduction in vortex lift encountered at supersonic speeds as the Mach cone approaches the leading edge. The latter is associated with the forward movement of the stagnation line which reduces the vortex strength until the sonic leading-edge case is reached and the leading-edge separation vortex vanishes—a phenomenon not accounted for by slender-wing theory.

The impact of the vortex flow on the lift-dependent drag parameter, $\Delta C_D/C_L^2$, as a function of C_L, is illustrated on the right of figure 8 for a Mach number of 2.0. It is seen that the suction analogy and experimental results are in good agreement and illustrate that the drag increase associated with the loss of leading-edge thrust due to leading-edge separation diminishes rather rapidly with increasing lift coefficient. This phenomenon is, of course, a result of the reduced angle of attack required for a given lift coefficient when vortex flow is present.

This initial supersonic study made it clear, as in the subsonic case, that the incremental drag reductions available through camber and twist for the subsonic-edge case are considerably less than predicted by methods which ignore the vortex-lift effect on the zero suction case (see ref. 24).

The above research was extended soon after by Fox and Lamar (ref. 25) who performed a theoretical and experimental study on a very slender wing which was within the Mach cone well into the hypersonic speed range.

Regarding landing and takeoff performance characteristics, Fox (ref. 26) applied the analogy to the prediction of ground effects and validated his theory with an experimental study.

Basic research on the use of spanwise blowing to augment the vortex-lift capability of moderately swept wings was performed by Campbell who reviewed this and other jet-powered vortex augmentation schemes in reference 22.

Stability Characteristics - Slender-wing aircraft differ from their non-slender counterparts in such characteristics as the high angles of attack they encounter and their low inertia in roll, for example. These, when combined with the non-linear vortex flow characteristics made it important to develop a knowledge of the stability characteristics.

Examples of two of the stability-related studies carried out early in the program are illustrated in figure 9. On the left is an example from the theoretical and experimental study performed by Boyden (ref. 27) in which he investigated both the steady-state and oscillatory roll damping of slender wings. He developed a method of extending the analogy to the steady-state roll case and, as shown, his theory accurately predicted the large vortex induced damping.

The overall longitudinal load distribution, which is related to the longitudinal stability and pitch damping, is shown on the right of figure 9. Snyder and Lamar (ref. 28) have shown that although the analogy does not provide detailed surface pressures it does provide an accurate prediction of the longitudinal distribution of lift which can be translated into pitching moment and pitch damping. Their results illustrated the strong trailing-edge effects which limit the usefulness of conical flow theories.

Other stability-related studies include the lateral stability research performed by Davenport and Huffman (refs. 29 and 30) which covered the subsonic, transonic, and supersonic speed regimes and the investigation of vortex asymmetry by Fox and Lamar (ref. 25).

Some Extensions of the Analogy - By the early 1970's, the suction analogy had been found to provide accurate predictions of the vortex flow characteristics of slender sharp-edge delta wings for a wide variety of aerodynamic performance and stability parameters and was being routinely applied throughout much of the aeronautical community. The Langley research program was then extended to include arbitrary planforms and round leading edges as illustrated in figure 10. In addition to the theoretical developments, an extensive parametric wind tunnel study was performed to evaluate the resulting methods.

The general approach for the arbitrary planform extensions is illustrated on the left of figure 10 for the cropped delta configuration. Briefly, the method developed accounts for the additional vortex lift over the aft portion of the wing by the two additional vortex lift terms. The first, ΔC_{L_V}, accounts for the downstream persistence of the leading-edge vortex (but no additional feeding), while the second, $C_{L_{vse}}$, is a result of the additional feeding of vorticity predicted from the attached flow edge singularity distribution along the side edge. The initial research in this area, performed by Lamar, and the extensions by Luckring cover the wide variety of planforms listed on the figure and the details of this research have been reviewed by Lamar and Luckring in reference 21.

Let us now turn from the sharp-edge cases to those with round edges where the separation is no longer fixed at the leading edge and the amount of leading-edge suction lost is a function of the location of the separation line. In the early studies, as illustrated on the right of figure 10, some variations in the measured vortex normal force, C_{N_V}, and in the remaining (or residual) leading-edge suction, C_{S_r}, were observed that led to the belief that there may be a "conservation of suction." Leading to that belief was the fact that available data on slender delta wings of various leading-edge shapes indicated that the sum of the vortex induced normal force and the remaining (or residual) portion of the leading-edge suction was equal to the theoretical attached flow leading-edge suction, $C_{S_{th}}$. The original Langley study, performed in 1974, was published by Kulfan (ref. 31), with permission, who used it to develop a prediction method. The research was continued by Henderson (ref. 32), who found the concept to hold for a variety of configurations. More recently, this concept has been used to develop a vortex flow prediction method for both subsonic and supersonic flow by Carlson and Mack (ref. 33).

AIRCRAFT CONFIGURATION RESEARCH

During the latter part of the 1960's as Air Force interest in a new lightweight highly maneuverable fighter was growing, Langley researchers expanded their aerodynamic research in several related areas. One area was the application of vortex lift to provide a lightweight approach to the high lift capability required for transonic maneuvering as well as takeoff and landing performance. Some of the conceptual configuration types studied in the vortex lift program are illustrated in figure 11. The two general wing types are characterized as "classical slender wings" and "hybrid wings."

The configurations utilizing classical slender wings are represented here by the conventional slender delta tailless type and the close-coupled canard delta. The "hybrid" wings combine attached flows and vortex flows in various combinations to provide additional degrees of multi-design-point capability. Two subclasses of hybrid wing concepts are illustrated, one which used vortex-lift strakes and is biased towards transonic maneuvering and the other a slender cranked wing biased towards supersonic cruise. For this paper, the review of the research program will be limited primarily to the vortex strake concept. A more complete review of the overall program can be found in reference 9.

Vortex-Lift Strakes

Two events contributed to the development of Langley's vortex-lift strake research. As a result of their canard-wing research and their basic research related to the vortex lift of slender wings, it began to appear to the Langley researchers that the favorable effect of the canard trailing vortex (resulting from the energizing effect its sidewash produced on the wing upper surface boundary layer near stall) might be extended to higher angles of attack by the highly stable leading-edge vortex flow of a slender lifting surface (see ref. 34). During the same general time period, the Northrop Company noted a favorable impact on the maximum lift of the F-5A due to a small flap actuator fairing that extended the wing-root leading edge. This spurred interest in the influence of inboard vortex flow and eventually led to the development of the YF-17.

As a result of the Langley and Northrop vortex interaction studies, plans began to be formulated by mid 1971 for an expansion of the Langley program to investigate the hybrid-wing approach with the slender lifting surfaces which became known at Langley as "vortex-lift maneuver strakes." The initial phase of the program reported by E. J. Ray et al. (ref. 35), which was performed in the Langley 7- by 10-Foot High-Speed Tunnel during the early fall of 1971, utilized the double balance technique to isolate the strake and wing loads and appears to be the first tests to clearly illustrate the magnitude of the favorable effect the strake vortex flow induces on the main wing panel flow at maneuvering conditions. Figure 12 illustrates the large overall lift increase produced by the strake at maneuvering conditions. Also shown is the direct lift carried by the strake-forebody and the incremental lift changes on the main wing panel. The total lift results illustrated the nonlinear character of the lift produced by the strakes which produces high levels of maneuver lift with essentially no increase in high-speed low-altitude gust response. The division of the lift produced by the addition of the strake illustrated the large lift increment produced on the main wing as the highly stable vortex from the strake reorganizes the flow and delays the stall on the outer panel.

This study also demonstrated the large drag reductions in the high lift range. Recognizing that the degree of flow control on the main wing would be a function of the wing design, tests were also made with segmented leading-edge flaps deflected to simulate a high-lift design condition. As was expected, the tests indicated that as the wing design is improved to delay separation on the main wing panel, the beneficial effects of the strake are delayed to increasingly higher angles of attack.

From these studies, it appeared that the vortex-lift strakes combined with variable wing camber in the form of programmed leading-edge flaps could provide a low structural weight approach for the high maneuverability levels desired by the Air Force.

The Lightweight Fighters

In the fall of 1971, representatives of the Fort Worth Division of General Dynamics visited Langley to discuss a problem related to their lightweight fighter design study (ref. 36). The design incorporated a lifting fuselage in the form of a wide, flattened, and expanding fuselage forebody that blended into the wing. The uncontrolled separation from the fuselage forebody for this design was creating stability and performance problems at maneuvering conditions. The Langley researchers suggested that the edge of the wide "lifting" forebody be sharpened to fix the separation line. In addition to controlling the forebody separation, this would increase the strength and stability of the vortex shed from the forebody, thereby increasing the vortex lift as well as stabilizing the high angle of attack flow field over the aircraft. After their own studies of the suggestion, General Dynamics included the vortex-lift strake in their design which became the well-known and highly maneuverable F-16.

By the mid 1970's, considerable interest in a supersonic cruise fighter aircraft had developed within the Air Force. Referred to as a "supercruiser," this fighter concept placed major emphasis on efficient supersonic cruise performance while maintaining respectable subsonic performance and maneuverability. As mentioned earlier, the strong emphasis on supersonic cruise tends to dictate a wing at the opposite end of the hybrid wing scale relative to the highly maneuverable transonic fighters just described. In this case it is now the main wing panel that is made slender to improve supersonic cruise performance as well as utilize the vortex lift. The cranked outer panel provides improved subsonic and transonic performance. An extensive research program was carried out, and the reader is referred to reference 9 and its cited references for details of the program. The concept eventually was applied in the F-16XL "derivative" aircraft, resulting in an excellent combination of reduced supersonic wave drag, controlled separation in the form of vortex lift, and low structural weight while maintaining the wing span desired for subsonic performance.

Photographs of these two hybrid-wing aircraft in flight are shown in figure 13.

FREE-VORTEX-SHEET THEORY

The vortex lift design application just discussed was aided considerably by the suction analogy. However, a considerable amount of wind tunnel testing was required and, as in the case of attached flows, there is a continuing need for refinements in the theoretical modeling of the real flow to keep pace with aircraft design requirements.

The need for a theoretical model of the complete, three-dimensional flow field was recognized early in the Langley research program and, as interest in vortex flows accelerated, the Boeing/LRC free-vortex-sheet method was developed. The Langley researchers worked closely with Boeing to define the applicational needs and evaluate the method during development. The initial development work and some early applications were described in a joint paper by Gloss and Johnson (ref. 37).

The Basic Formulation

A schematic of the free-vortex-sheet model (FVS) is shown in the left of figure 14. The vortex sheets are modeled with biquadradically varying doublet panels representing: (1) the free sheet shed from the separation line, (2) the fed sheet which is a simplified model of the vortex core region, (3) a higher order near wake, and (4) a far (or trailing) wake. Neither the shape of these three-dimensional sheets nor strength of the doublet distribution is known a priori resulting in a nonlinear problem requiring iteration schemes. What is essentially the current state of the free-vortex-sheet theory is described by Johnson et al. (ref. 38).

In addition to working closely with Boeing during the development, Langley researchers have made comprehensive validation and application studies, some of which have been reviewed by Luckring, Schoonover, and Frink, in reference 39. They describe their investigation of convergence techniques for both the wing flow and the near-wake flow as a means of reducing computational cost and present several examples of practical applications. Based on these and other studies, it appears that the basic version of the theory has provided the most accurate and versatile inviscid approach available for establishing the complete three-dimensional flow field and surface pressure distributions for arbitrary configurations throughout the subsonic flow regime. A review of the convergence capabilities of the free-vortex-sheet theory and a survey of its applications are covered in this conference by Luckring et al. (ref. 40).

Some Recent Extensions

In addition, the free-vortex-sheet theory appears to offer an excellent inviscid flow model to which various viscous effects can be added, and two of these type extensions are illustrated by the cross-sectional cuts presented on the right of figure 14.

The first deals with the stability of the primary vortex and its influence on vortex breakdown, a phenomenon that is often the primary factor in limiting the maximum lift attainable. To include the vortex breakdown in the basic theory Luckring (refs. 41 and 42) has coupled Navier-Stokes inner and outer core regions with the inviscid free sheet and investigated various vortex instability criteria. His resulting theoretical model appears to accurately include the important effect of the pressure gradient associated with the trailing-edge Kutta condition on the vortex breakdown.

The second viscous flow addition to the free-vortex-sheet theory, shown in figure 14, is the inclusion of the secondary separation which occurs when the boundary layer on the upper surface, which is swept towards the leading edge by the primary vortex flow, separates under the influence of the adverse spanwise pressure gradient outboard of the primary vortex. The resulting flow can include secondary and tertiary vortices and produces important redistributions of the surface pressures. Two approaches to the inclusion of the secondary separation in the theory are described in detail in other papers presented during this conference (refs. 43 and 44) and, therefore, will not be discussed in this paper.

Euler and Navier-Stokes solutions of the leading-edge vortex flow phenomenon, while not as yet being as generally applied to design and analysis projects as are the suction analogy and free-vortex-sheet theory, appear to offer extended capabilities for the future. Research on these methods is included in this conference (refs. 45 and 46) and will not be reviewed here.

SOME CURRENT CHALLENGES

Since reviews of the progress in the development and application free-vortex-sheet theory will be presented in other papers in this conference, I have elected to use the theory to highlight a few computational and experimental challenges and to encourage a coordinated development of the various vortex flow theories and a close cooperation between theoreticians and experimentalists. The challenges discussed here will be confined to a few incompressible flow examples. However, some compressible flow challenges will be discussed in the overview paper by Campbell and Osborn (ref. 47) and many other undoubtedly surfaced during this conference.

Predicted Flow Regimes

Figure 15 illustrates the free-vortex sheet theory prediction of three vortex flow regimes encountered on sharp-edged delta wings. The boundaries of the regimes are presented as a function of leading-edge sweep angle and angle of attack. One boundary is associated with the vortex stability as predicted by Luckring's addition of the viscous core regions to the free-vortex-sheet theory and is defined by the critical swirl condition (see sketch) of tangential velocity equal to axial velocity at the trailing edge. Above this boundary, the vortex would be expected to be unstable. It is interesting that the data of Wentz and Kohlman (ref. 48), defining vortex breakdown at the trailing edge, appears to substantiate the theory. Also shown is the experimental buffet onset condition established by Boyden and Johnson (ref. 49). A computational challenge in this flow regime might be the application of the theory to develop wing design criteria related to the delay of vortex breakdown to provide extended lift capability.

The other boundary shown in figure 15 is derived from the completely inviscid version of the free-vortex-sheet theory and establishes the angle of attack above which the vortex interactions, or crowding, cause the core paths to begin to diverge from each other laterally as illustrated in the sketch. Results are shown (for two wing thickness ratios) at the 40% longitudinal station and are typical of other stations. It will be noted that the experimental angle of attack corresponding to the onset of vortex asymmetry reported by Fox and Lamar (ref. 25) for a sharp-edge delta wing of aspect ratio 0.25 lies in the diverging core regime. Secondary separation effects would be expected to influence the boundary and represent a challenge related to the use of the free-vortex-sheet theory in studying certain aspects of the development of vortex asymmetry. It must be recognized, however, that there are, in all probability, other viscous-related effects that limit the core divergence and influence the criteria for stable asymmetric vortex flow.

Overall Vortex Lift Effects

The influence of the above flow regimes on the overall lift characteristics as predicted by the free-vortex-sheet theory is illustrated in figure 16 for two slender sharp-edged delta wings having leading-edge sweep angles of 70° and 80°. The free-vortex-sheet solutions are shown by the solid lines, and the dashed lines represent solutions by the suction analogy. Experimental data (ref. 48) are shown by the symbols. The lift calculated by the suction analogy is believed to provide the "upper bound" of lift for conditions where no losses associated with vortex crowding or vortex breakdown are encountered. The shaded region between the two theories represents the angle-of-attack range where the two theories depart. It is believed that this comparison illustrates the vortex crowding effect, discussed above, which hinders flow reattachment and results in incomplete recovery of suction as vortex normal force. The angle of attack corresponding to the diverging core boundary calculated by the free-vortex-sheet theory is shown by the solid arrow for both wings and appears to reasonably define the onset of incomplete suction recovery.

For the 70°-delta wing, the suction analogy and the free-vortex-sheet theory are in excellent agreement up to about an angle of attack of 40° and both agree with experiment until vortex breakdown is encountered. However, for the 80°-delta wing, the free-vortex-sheet solutions begin to show lift losses in the vicinity of 20°.

Also shown (by the half-solid arrows) are the calculated values of the angle of attack at which the critical swirl condition is predicted, and it is seen to be in reasonable agreement with the maximum lift coefficient which has been shown experimentally (ref. 48) to correspond closely to the vortex breakdown condition.

With regard to the maximum lift, the free-vortex sheet theory results shown for the 80° delta indicate that little increase would be expected by eliminating vortex breakdown. However for the 70° delta, which is currently of more practical interest for aircraft, the free-vortex sheet solutions indicate that large increases in maximum lift might be attainable by delaying vortex breakdown. As mentioned in the previous section, this offers a challenge to use the theory to establish wing warp and planform shaping that are more conducive to vortex stability. However, additional effects such as possible shocks or vacuum limits at the higher lifts must, of course, be considered.

With the comparisons of the suction analogy and the free-vortex sheet solutions offering a means of demonstrating the magnitude of various "real flow" effects, it is highly recommended that Euler and Navier-Stokes' studies of vortex lift be coordinated with the free-vortex-sheet studies as well as with those experimental studies which are designed to provide detailed knowledge of the "real flow."

Secondary Separation

It has been reasonably well established that the overall lift of thin, sharp-edge, slender delta wings, of interest here, is relatively insensitive to Reynolds number. For this class of wings, where the primary separation is fixed at the sharp leading edge for any non-zero angle of attack regardless of the state of the boundary layer approaching the sharp edge, this is not totally unexpected. However, flow details associated with the state of the boundary layer on the wing upper surface can cause rather large variations in the local pressure distributions through their effects on the location of secondary separation lines. These well-known secondary separations occur when the flow under the primary vortex is swept toward the wing leading edge, by the action of the vortex, and encounters the adverse spanwise pressure gradient near the leading edge. This results in a secondary vortex having vorticity of the opposite sign. In general, the effects induced by the secondary separation have been envisioned using the vortical effects of the secondary vortex. However, an alternate modeling approach has been used, with reasonable success, by Blom et al., and Wai et al. (refs. 44 and 50) who assumed that "the secondary vortices are so embedded in the boundary layer that their displacement effects dominate over their vortical effects." Regardless of the various modeling details, the experimental secondary-separation effects described below will be referred to as a "secondary vortex."

The secondary-separation characteristics are, of course, highly dependent upon the local state of the boundary layer and, therefore, on the Reynolds number and longitudinal station. This is illustrated in figure 17 using experimental, transition free, data from reference 51 for a 76° sharp-edge delta wing at an angle of attack of 25°. On the left is presented the spanwise variation of the upper surface pressure distribution at the mid longitudinal station for Reynolds numbers, based on root chord, of 1.6×10^{6} and 6.4×10^{6}. Also shown is the inviscid pressure distribution predicted by the free-vortex-sheet theory (FVS) including the effects of wing thickness but assuming no secondary vortex. The low Reynolds number data represent a laminar secondary separation condition and the well-known large reduction in suction pressures in the region of the primary vortex, accompanied by large increases in suction pressures in the region of the secondary vortex, is clearly evident. The higher Reynolds number data illustrate the turbulent separation case in which the secondary vortex is reduced in strength and its formation delayed to a more outboard position. The pressure distribution now approaches the theory more closely in the region of the primary vortex but still shows important effects near the leading edge which is particularly important for cambered wings or vortex flaps.

The right-hand portion of figure 17 illustrates the strong non-conical effects on the pressure distributions along mid semispan ray, $y/s = 0.5$, for both the inviscid theory and the experimental results. The most complete data were obtained at a Reynolds number of 3.2×10^{6} and clearly illustrate the transition from a laminar secondary separation to a turbulent separation in the region of $x/c_o = 0.35$. Although the data for the other Reynolds numbers are incomplete, they appear consistent with the expectation that transition would occur downstream for the lower Reynolds number and upstream for the higher Reynolds number. Regarding the inviscid theory results from the free-vortex-sheet method, it appears that the theory may provide a reasonably accurate prediction of the pressures in this region of the wing as long as the turbulent secondary separation has its origin near the wing apex. It is of interest, however, that while the experimental results approach the theory at the high Reynolds number they do not do so in a monotonical fashion as will be discussed below.

To more clearly indicate the variation of the peak suction pressure coefficients under the primary vortex with Reynolds number, data taken from reference 51 have been analyzed in a somewhat different fashion and presented in figure 18 as a function of Reynolds number for both the $x/c_o = .25$ and $x/c_o = .50$ longitudinal stations. Also shown are the peak suction values measured under the secondary vortex.

Looking first at the $x/c_o = .25$ station, the experimental data indicate a rapid increase in peak suction level under the primary vortex with increasing Reynolds number as the cross flow under the vortex becomes turbulent and reduces the impact of the secondary separation. To provide a possible high Reynolds number asymptote for this suction peak, the value obtained from the inviscid free-vortex-sheet theory is also shown. Although it appears that the data may be approaching the inviscid theory monotonically, evidence to be discussed subsequently, relative to the $x/c_o = .50$ station, indicates a more complicated situation.

Also shown in the left of figure 18 is the variation of the experimental peak suction pressure under the secondary vortex. In this case when turbulent secondary separation occurs, there is a rapid decrease in the peak suction pressure magnitude.

Turning to the $x/c_o = .50$ case on the right of figure 18, it is seen that, as discussed earlier, the transition to a turbulent secondary separation and the accompanying rapid increase in the suction pressure under the primary vortex occur at a lower value of Reynolds number than for the upstream station. However, it is important to note that as Reynolds number increases further, instead of approaching the inviscid theory monotonically, there is a decrease in the magnitude of the peak suction pressure coefficient under the primary vortex. Accompanying this variation is an increase in the magnitude of the peak suction pressure coefficient under the secondary vortex. The reason for this somewhat unexpected variation with Reynolds number is not completely apparent and more complete data extended to higher Reynolds numbers would be of great value. The fact that the decrease in the peak suction pressure coefficient under the primary vortex at the $X/c_o = .50$ station occurs in the same Reynolds number range as the increase at the $x/c = .25$ station may be an indication that as the transition to turbulent secondary separation moves forward, the resulting effects induced downstream may cause changes in the primary vortex strength such that no station asymptotes the inviscid theory value until transition occurs very near the wing apex. This would appear to be somewhat consistent with the observed insensitivity of the total lift to Reynolds number and possibly consistent with the apparent "conservation of suction" related to the suction analogy.

The above observations imply an important challenge in the development and evaluation of advanced theoretical methods as well as a challenge to extend the pressure data to higher Reynolds numbers.

A New Aerodynamic Facility

During 1984, the Langley Research Center placed in operation a new, high Reynolds number, transonic wind tunnel. This tunnel, the National Transonic Facility (NTF), applies the cryogenic, pressurized, wind tunnel principle to provide the very high Reynolds numbers required to match the full-scale viscous effects encountered with modern air vehicles. The cryogenic technology development and the construction and operation of the pilot facilities were performed by Langley researchers and technicians (see refs. 52 and 53), and the basic design of the NTF was carried out in a Project Office staffed by Langley personnel (see ref. 54). The performance envelope of the NTF (ref. 55) is shown in figure 19, and the degree to which the NTF extends the Reynolds number capability beyond the composite envelope of all other operational tunnels in the free world is readily apparent. This capability is of considerable importance to vortex lift research since strong viscous effects can occur at design conditions as well as off-design conditions.

As seen by the sketches in figure 19, the vortex lift research currently planned for the NTF includes the classical slender delta wing and the two classes of hybrid wings discussed earlier.

The 65^{o} delta wing model incorporates an interchangeable leading edge to allow the study of leading-edge radius which, of course, has a strong effect on leading-edge separation characteristics, including the Reynolds number dependency. The model contains a large number of surface pressure orifices and is expected to provide considerable insight regarding the secondary separation characteristics in the full-scale Ryenolds number range as well as evidence related to the question of the existence of a "conservation of suction" phenomenon. The unique capabilities of the NTF will also be utilized to study possible effects of condensation in the core on vortex bursting.

The vortex lift strake configuration utilizes the double-balance system to isolate overall loads and component interactions as well as wing and fuselage pressures. Although not as highly instrumented for detailed vortex flow research, the slender, cranked, wing configuration should offer valuable overall force and moment information with regard to vortex flow.

Additional details of the NTF models and research program can be found in references 56 and 57, and it is highly recommended that researchers involved in the development and evaluation of advanced vortex flow theoretical methods consider this program in the selection of configurations to be modeled theoretically.

CONCLUDING REMARKS

By way of conclusion, I would like to emphasize that, by design or not, vortex flow can be encountered on slender-wing aircraft at many points within their operational envelope. Since this flow influences both the aerodynamic and structural design, it is important that continued improvements in design and analysis theories be developed to meet the increasingly stringent design requirements. Towards this end, it appears highly desirable that a strong interaction be developed between those developing theoretical methods such as, for example, the free-vortex-sheet, Euler, and Navier-Stokes. Since viscous effects such as secondary separation can strongly influence the design of cambered leading edges and vortex flaps, for example, it is essential that both laminar and high Reynolds number turbulent secondary separation capabilities be included in the theories. In relation to the influence of viscous effects, it is important that the experimental capabilities of the National Transonic Facility be utilized and that a close coordination between the basic and applied theoreticians and the experimentalists be developed.

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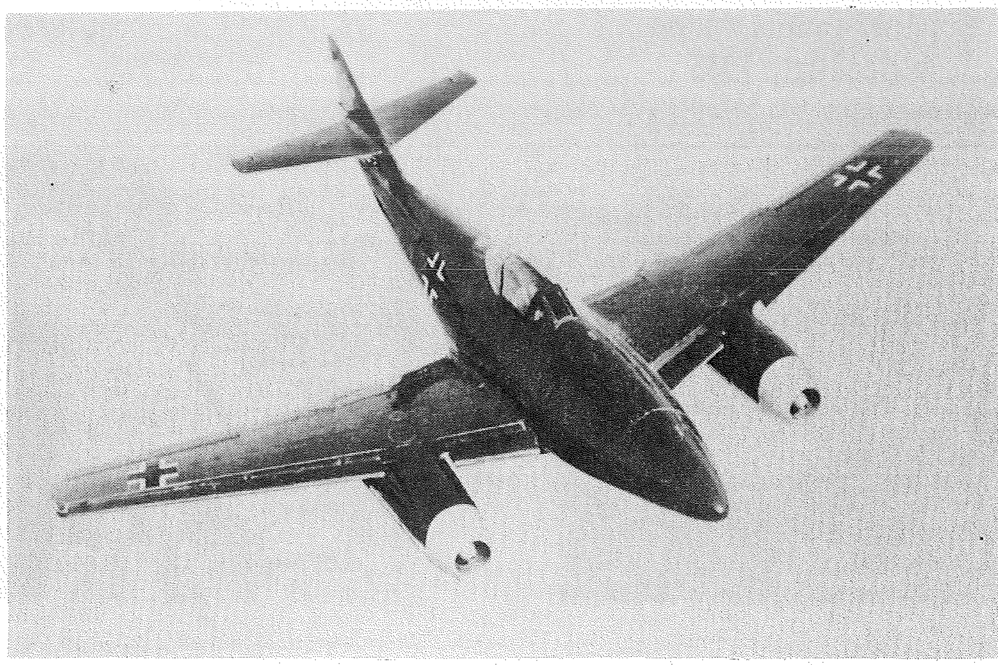


Figure 1. The Messerchmidt ME 262 - the first swept-wing jet aircraft.

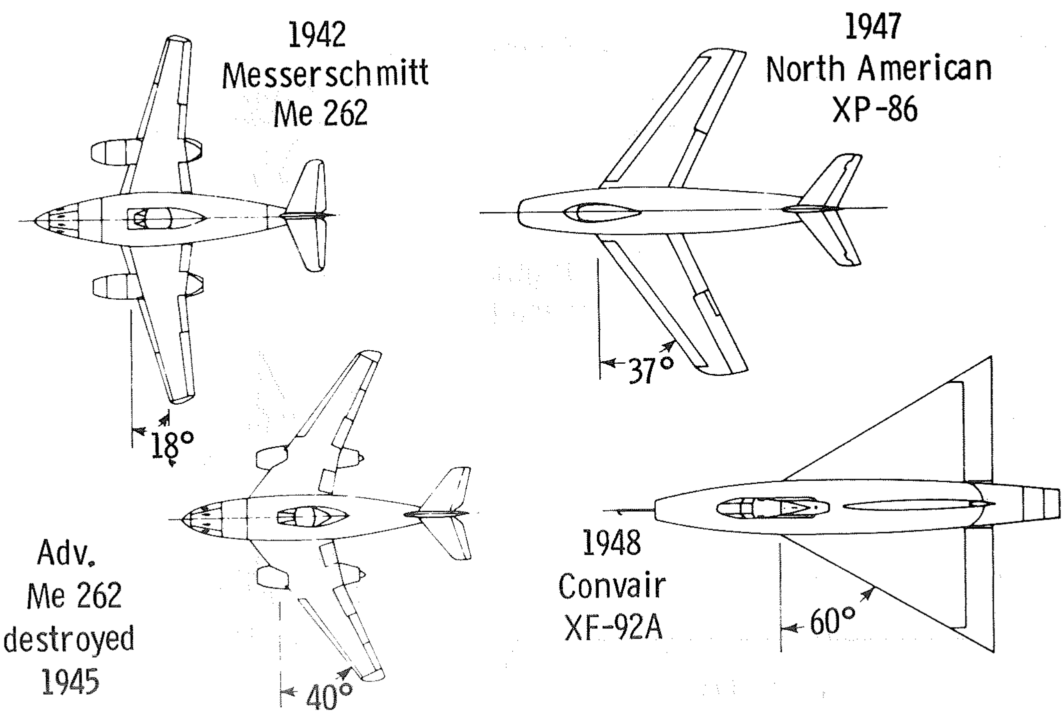


Figure 2. The trend towards slender-wing aircraft.

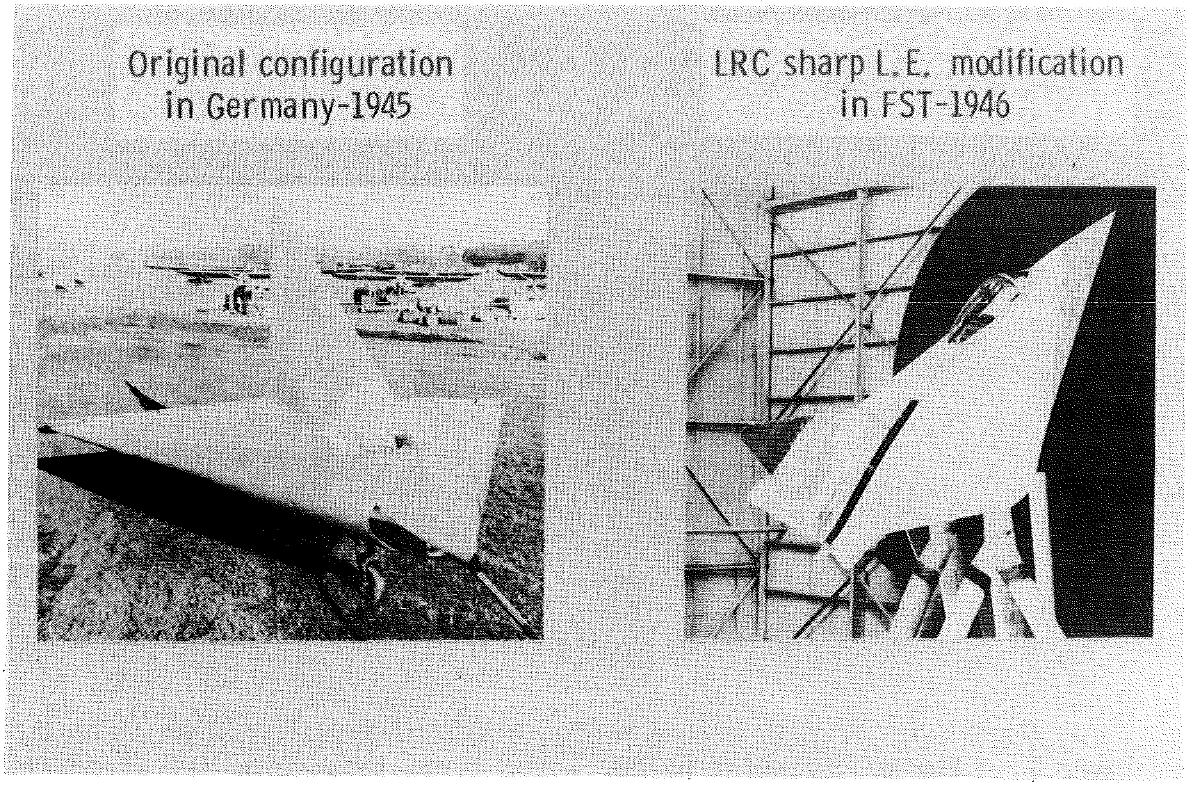


Figure 3. The Lippisch DM-1 test glider.

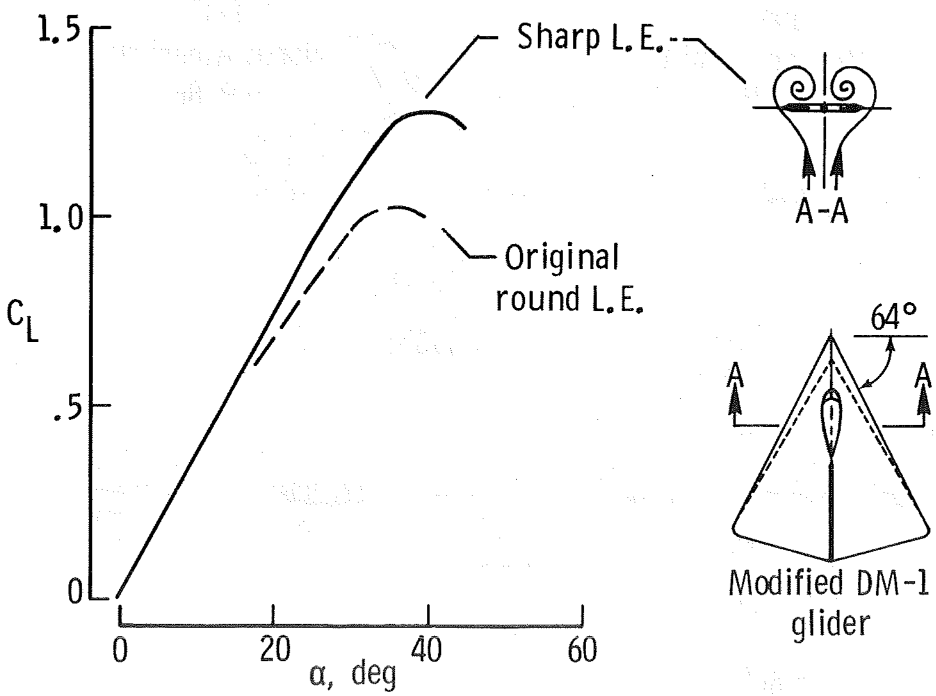


Figure 4. The first study of the leading-edge vortex occurred at Langley in 1946.

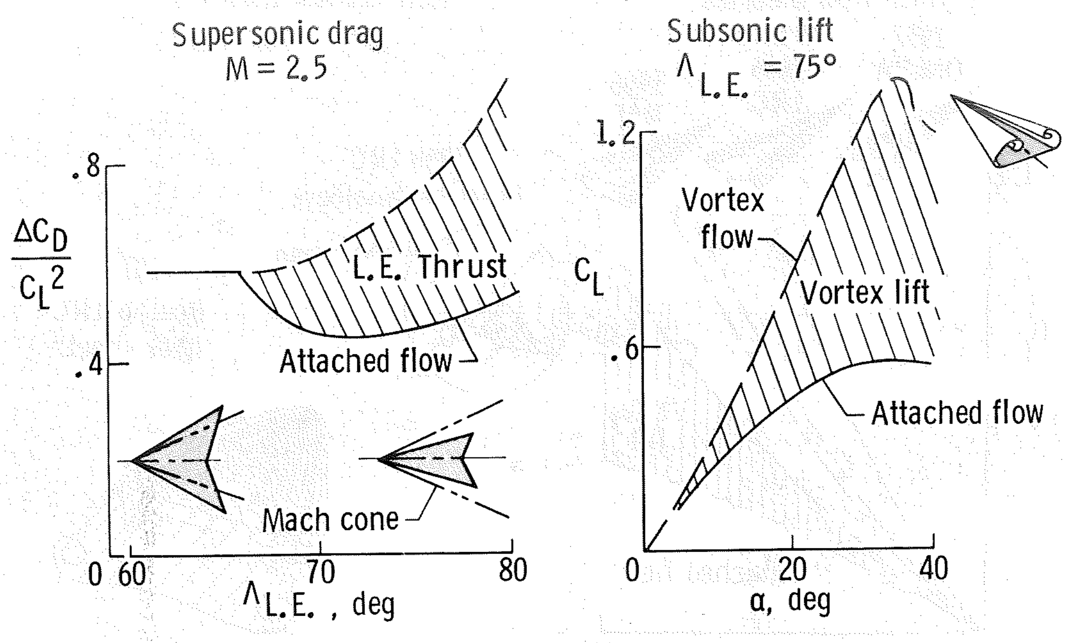


Figure 5. Some slender wing aerodynamic characteristics.

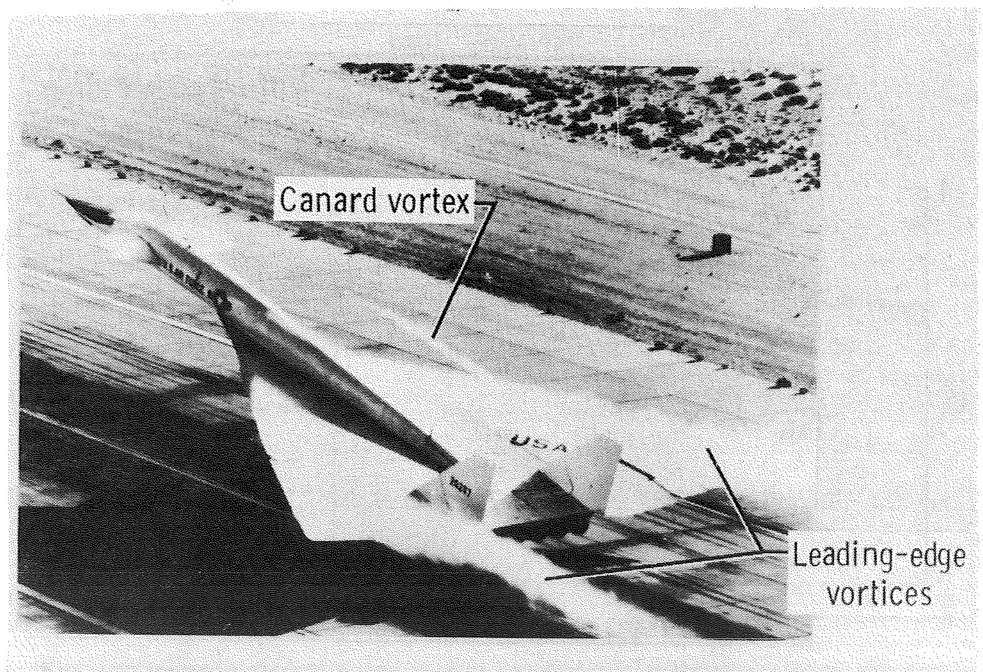


Figure 6. Illustration of the leading-edge vortices on the B-70.

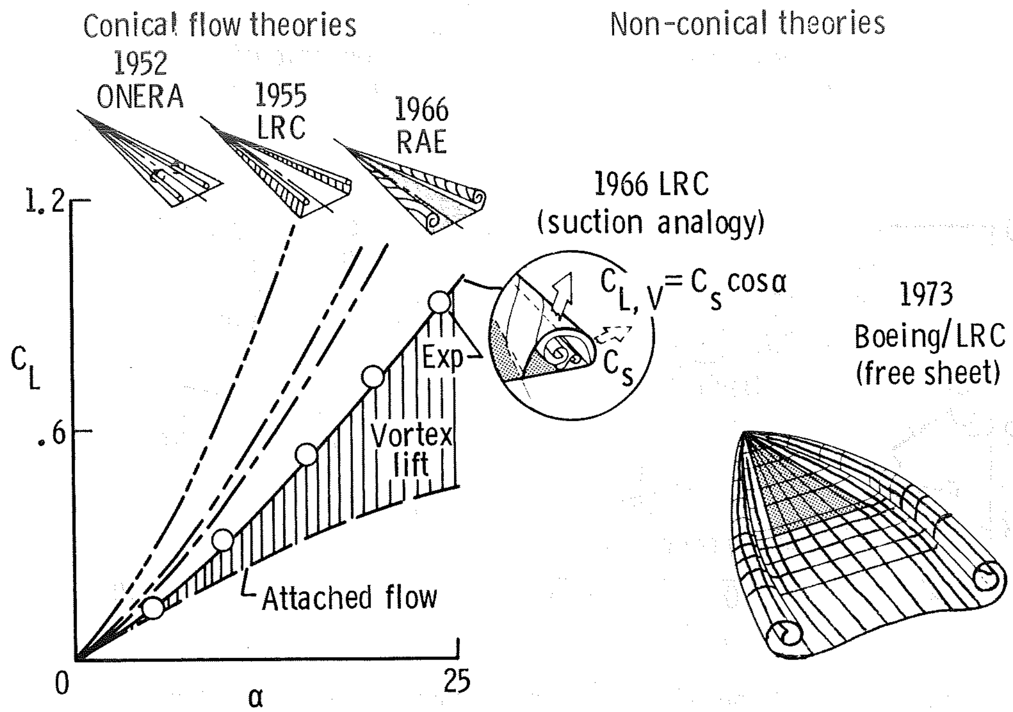


Figure 7. Chronology of vortex-lift theory development.

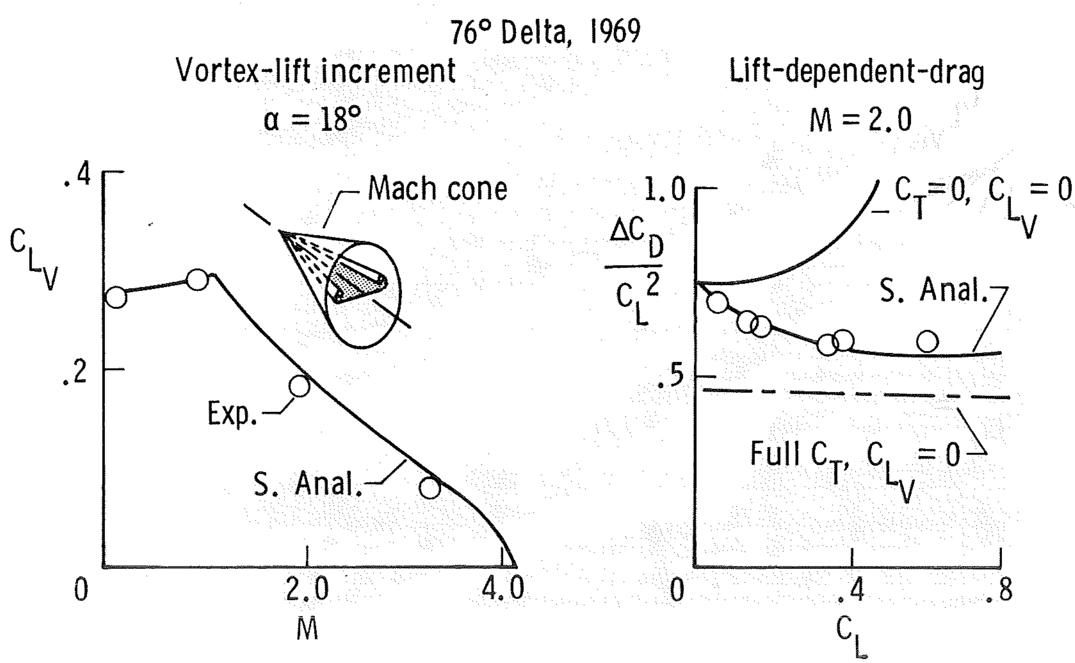


Figure 8. Prediction of some compressibility effects by the suction analogy.

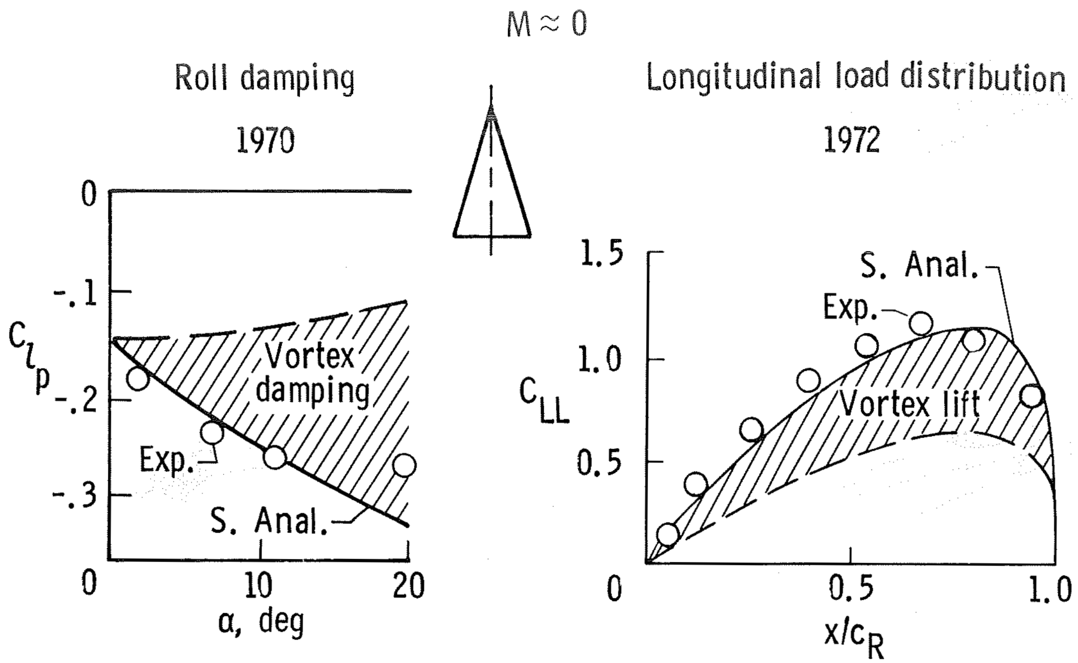
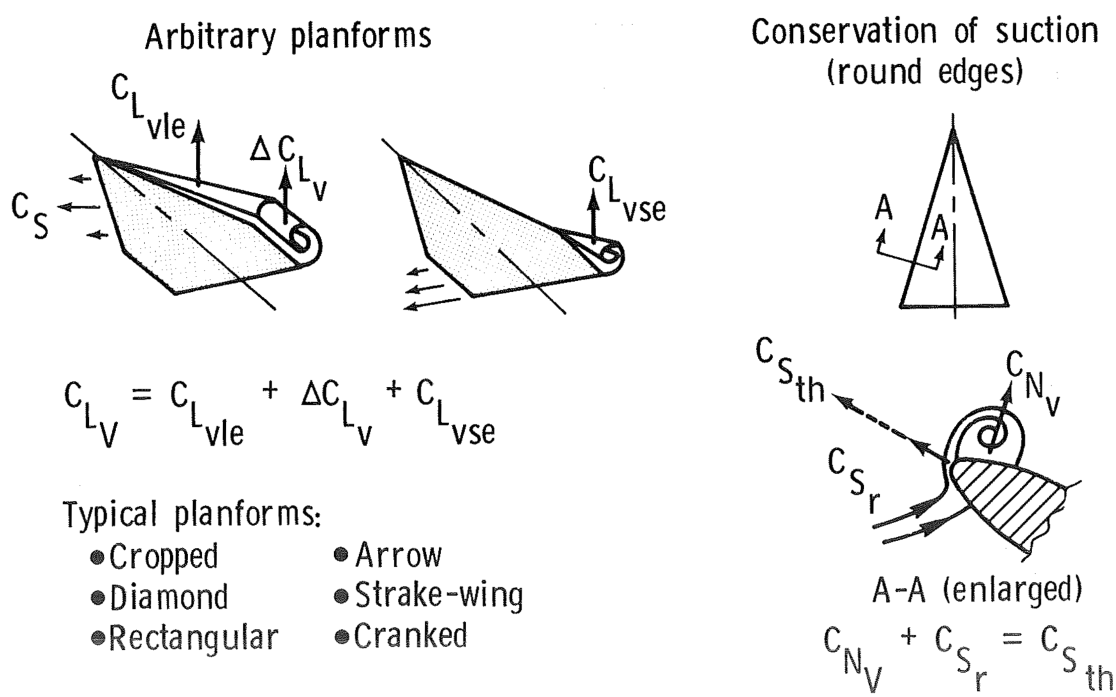


Figure 9. Prediction of some stability related parameters by the suction analogy.



$$C_{L_V} = C_{L_{vle}} + \Delta C_{L_v} + C_{L_{vse}}$$

$$C_{N_V} + C_{S_r} = C_{S_{th}}$$

Figure 10. Some extensions of the suction analogy made in 1974.

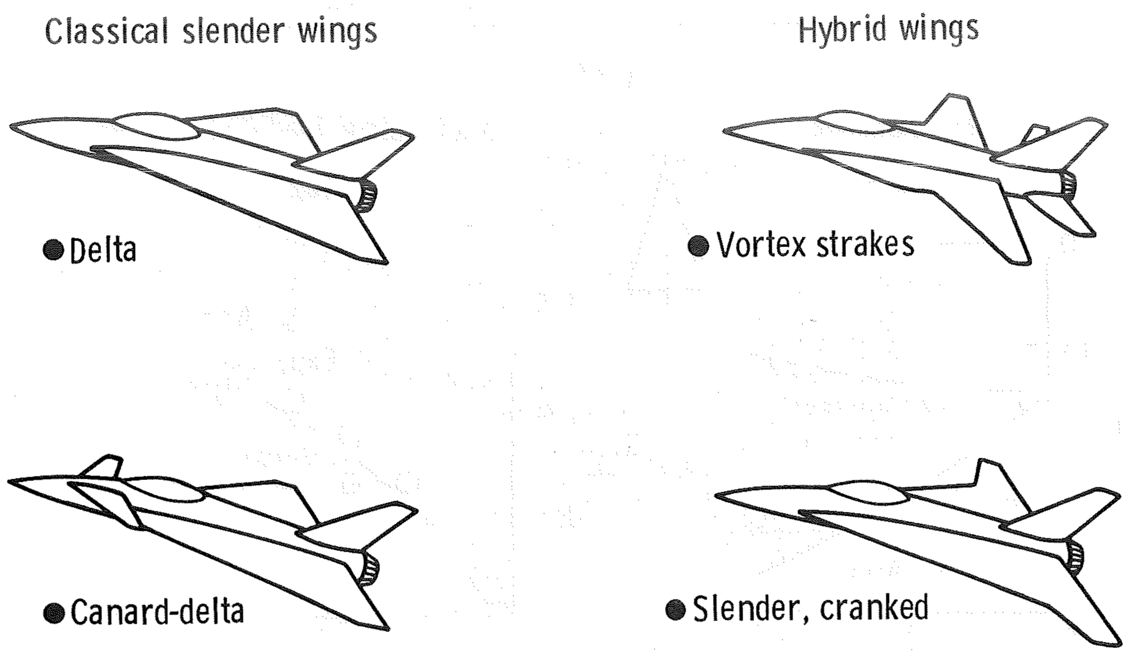


Figure 11. Some design applications of vortex lift.

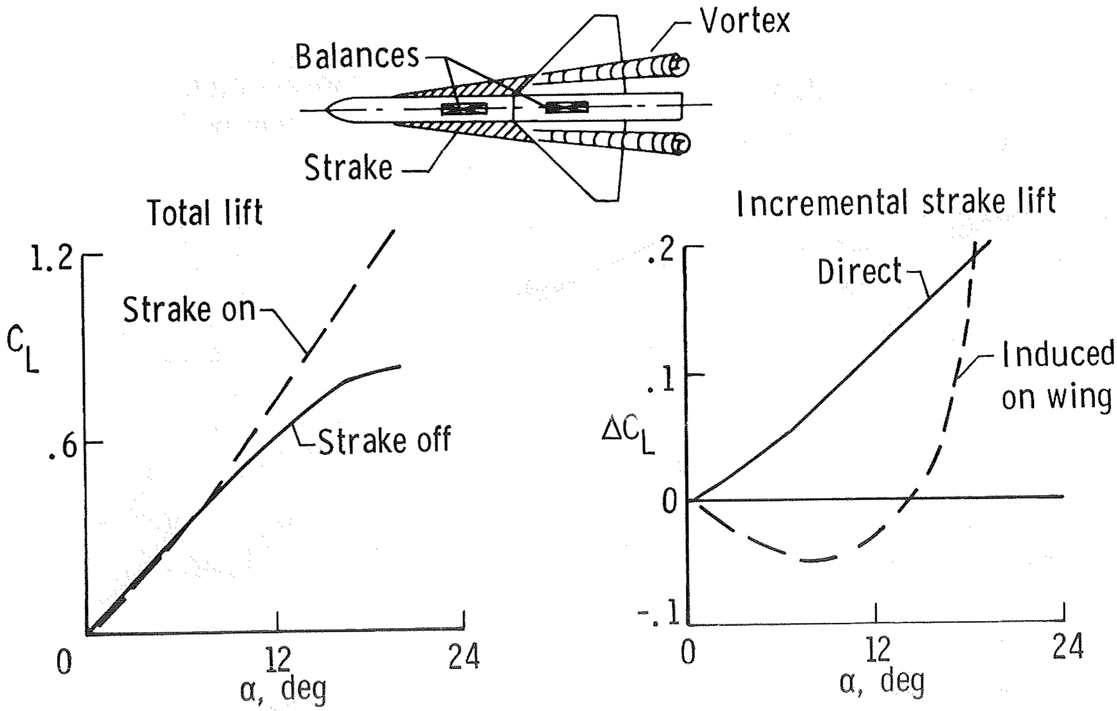


Figure 12. Effects of vortex-strakes on lift.

Figure 13. Examples of hybrid-wing aircraft.

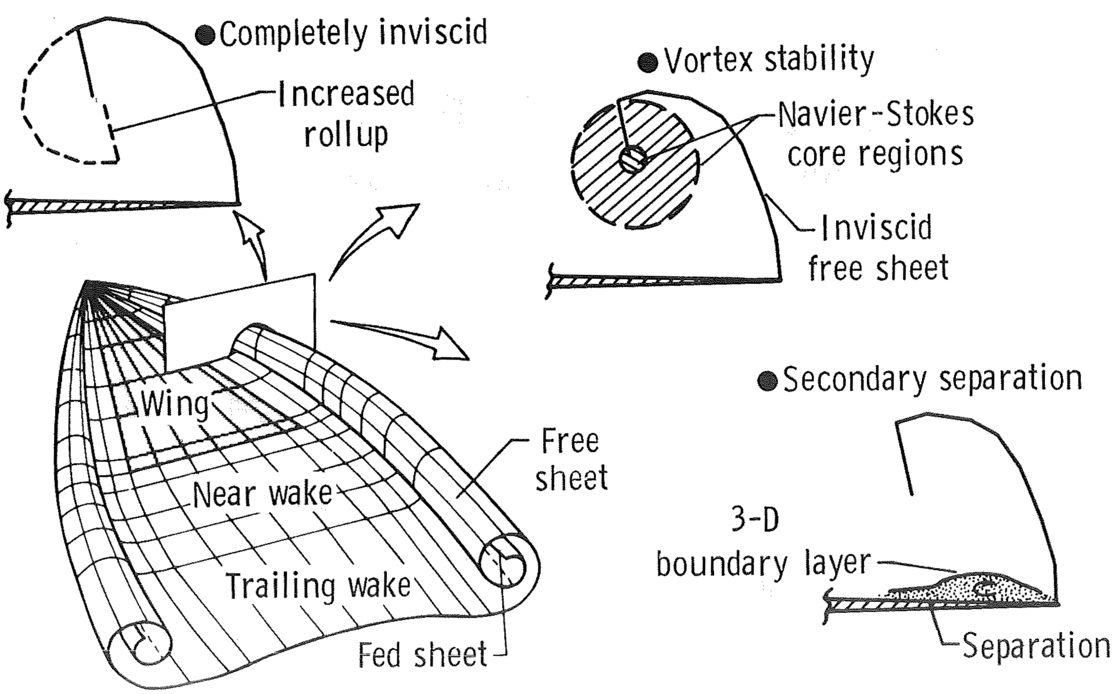


Figure 14. Some extensions of the free-vortex-sheet theory.

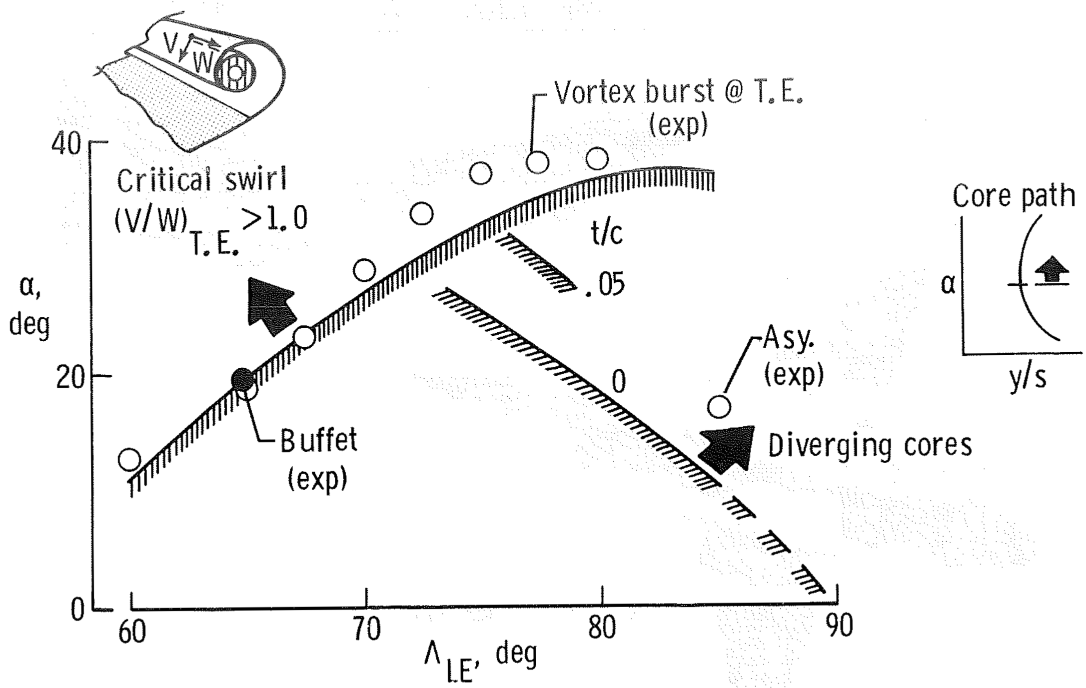


Figure 15. Prediction of vortex flow regimes by the free-vortex-sheet theory.

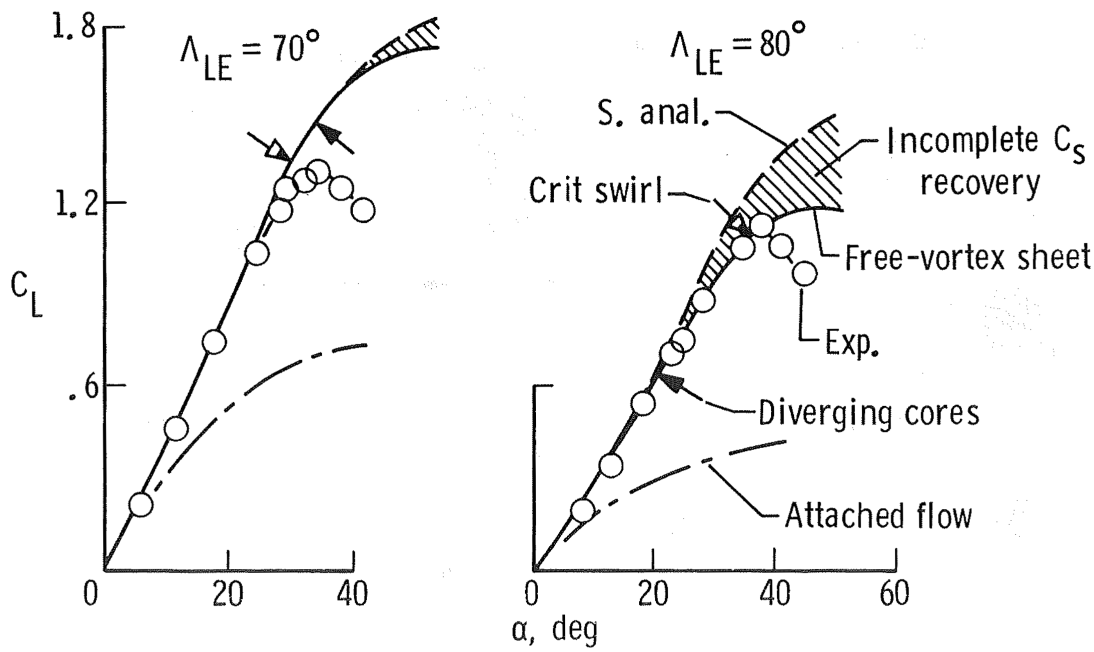


Figure 16. Influence of flow regimes on vortex lift of sharp-edge delta wings.

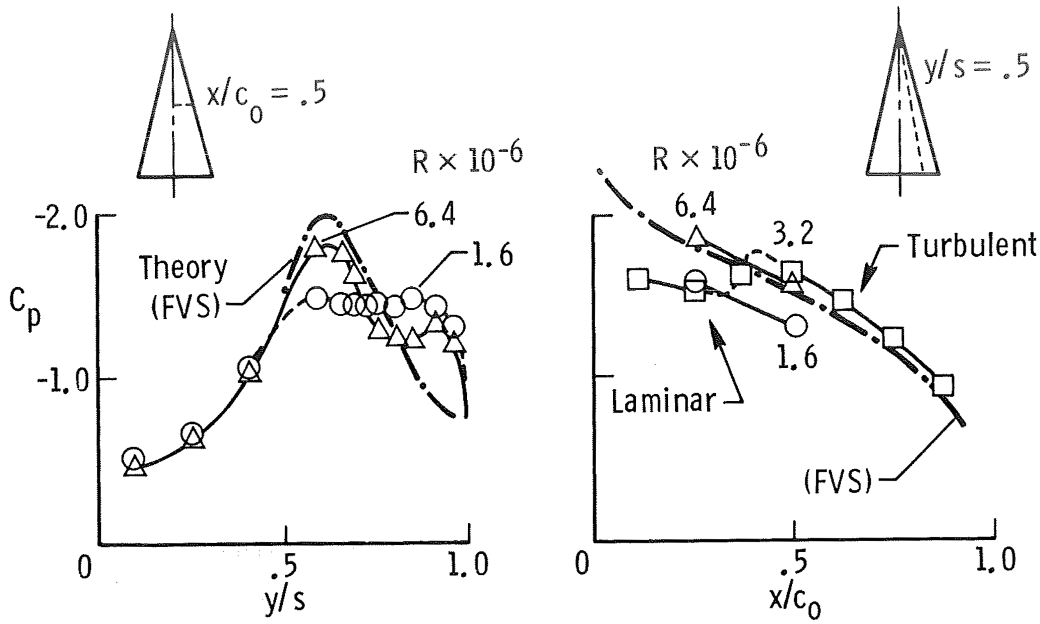


Figure 17. Effects of secondary separation on upper surface pressures. 76° delta at $\alpha = 25°$.

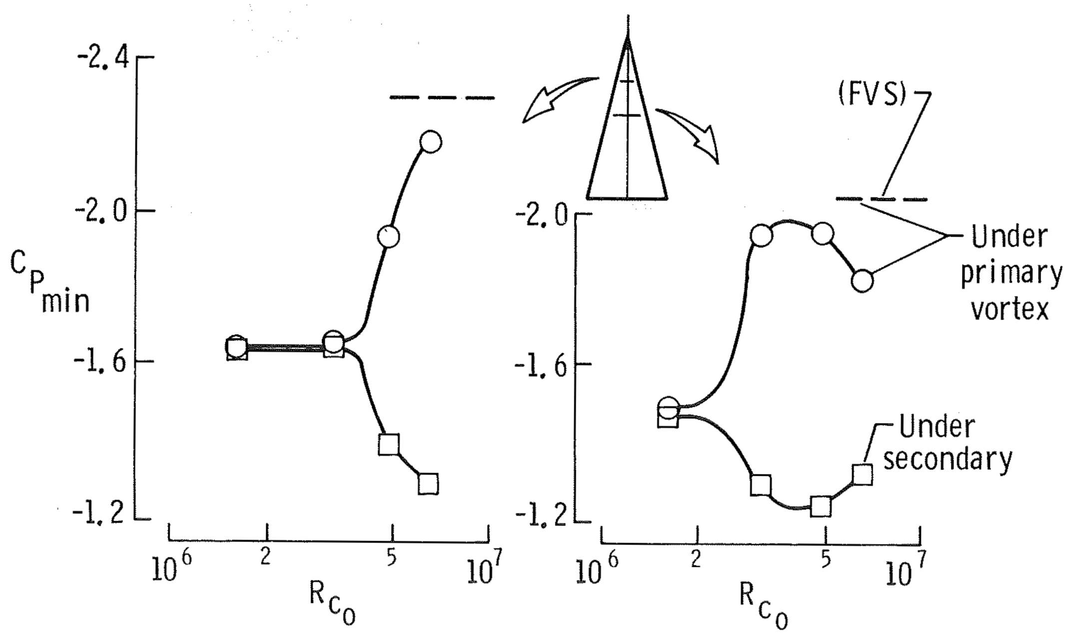


Figure 18. Effects of Reynolds number on upper surface minimum pressures. 76° delta at $\alpha = 25°$.

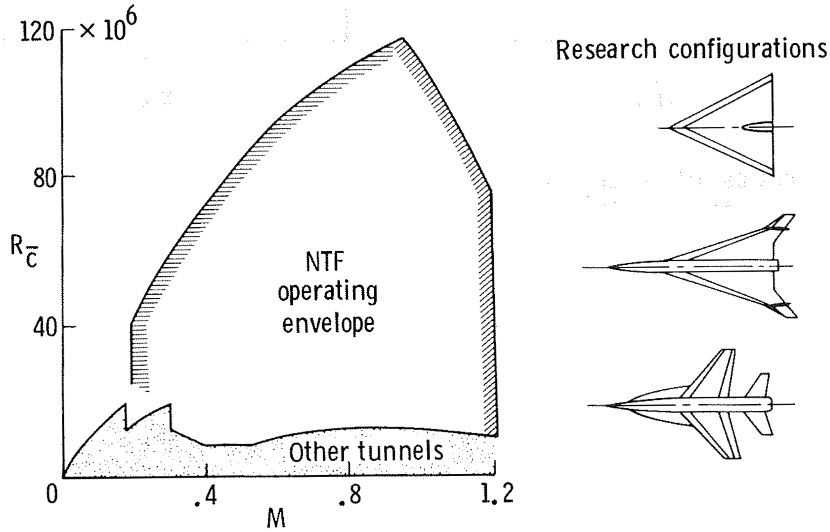


Figure 19. Reynolds number studies of vortex flow in the National Transonic Facility.