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AEROELASTICITY AT THE NASA LANGLEY RESEARCH CENTER - RECENT PROGRESS, NEW CHALLENGES

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INTRODUCTION

"It is but repeating a truism to open this discussion with the statement that aeroelastic phenomena occupy a principal role in determining the flight characteristics of modern aircraft. The demands of higher speeds and increased structural efficiency have resulted in aircraft of greater relative flexibility. Since airframe distortions are inevitably accompanied by changes in the aerodynamic loading, flexible aircraft display a variety of static and dynamic behaviorisms which require design control, or which must be used to design advantage."

Martin Goland, then vice-president of Southwest Research Institute, made these observations almost thirty years ago (ref. 1). More than ever this role that aeroelasticity plays in the design of aerospace vehicles is becoming critically important to the success of innovative (some might say bizzare) configurations required to eke out the greatest possible performance for an increasingly varied spectrum of missions. Almost from the inception of powered flight, the aeroelastic characteristics of a design have been considered on the negative side of the ledger - i.e., "wing divergence," the "flutter penalty", etc. From that point of view, great strides have been made. The "avoidance" of aeroelastic problems has been more successful due to increased knowledge of the aeroelastic phenomena, better analytical and experimental prediction methods, the "computer revolution," etc. Some even hold aeroelasticity, from that point of view, to be a "mature discipline." In reality, a revolution is occurring in the perception of aeroelasticity - from that of a problem child to be dealt with to that of a knight in shining armor capable of wresting increased performance from configurations undreamed of a few years ago. This is made possible by advances in composite material technology, active controls technology, significantly increased computational capability, and the recognition of the need for a close working alliance between the aerodynamics, structures, and stability and control specialists very early in the design process (see, for example, references 2-9). With this approach we see desired flexibilities being designed into new configurations and naturally unstable designs being accepted on the premise that "active controls" will make things right (see, for example, references 10-15).

The aeroelastician therefore is finding himself "in a whole new ball game." As soon as he began to feel good about the progress that had been made in understanding and predicting unsteady airloads and the aeroelastic behavior of a variety of configurations tailored to specific flow fields (i.e. sharp leading edge, swept wings for supersonic flight, blunter airfoils for subsonic flight, attached flow for most of the flight regime) he is confronted with blended wing-bodies, relatively blunt high-speed airfoils, closely-coupled canard/wings, significant nonlinear influences of transonic flow as well as separated and vortex flow, large amplitude motions, conformal stores, increased structural flexibility, aerodynamically and structurally interfering surfaces, near coincident "rigid-body" and structural natural vibration frequencies, and fast-acting controls that can resonant-couple with everything from the wings to the pilot's eyeballs. The challenges and opportunities seem almost endless.

Given this state it seems appropriate to review recent progress in this area particularly at the NASA Langley Research Center (LaRC) to look at the questions answered and questions raised, and to attempt to define appropriate research emphases needed in the near future and beyond. The paper is focused primarily on the NASA Langley Research Center program because Langley is the "lead" NASA center for aerospace structures research, and essentially is the only one working in depth in the area of aeroelasticity. Although there is a significant level of effort and important contributions have been made in rotorcraft aeroelasticity at the LaRC, this area will be excluded to keep the discussion to a tractable length.

First, historical trends in aeroelasticity will be reviewed broadly in terms of technology and staffing particularly at the LaRC. Then selected studies of the Loads and Aeroelasticity Division at LaRC and others over the past three years will be presented with attention paid to unresolved questions. Finally, based on the results of these studies and on perceptions of design trends and operational requirements, future research needs in aeroelasticity will be discussed.

HISTORICAL TRENDS

The Technology

Aeroelasticity (the study of the static and dynamic response of an elastic structure to aerodynamic forces) and in particular flutter, a potentially castastrophic instability involving the interaction of aerodynamics and structural deformations, including inertial effects has influenced the evolution of aircraft since the earliest days of flight. There are several excellent survey papers which trace the recognition, study, and treatment of aeroelastic phenomena (in particular, flutter) from the turn of the century to the early eighties (refs. 16-18 for example). Reference 16traces some of the more important conceptual developments relating to the understanding and prevention of flutter from the Wright Brothers encounters with aeroelasticity to the rise in the importance of compressibility effects in the early fifties. Reference 17 traces the "first fifty years of aeroelasticity" to the late seventies, emphasizing British developments. Reference 18 covers experimental studies in the Langley Research Center Transonic Dynamics Tunnel (TDT) in the sixties and seventies.

A review of reference 16, especially, leads one to conclude again that "necessity is the mother of invention". Time and again it was the damage or crash of an airplane due to flutter that spurred efforts to understand, predict, and control the phenomena. Reference 16 cites much of the activity in unsteady aerodynamic theory development which took place in the thirties apparently in an ad hoc manner, and points out that even then compressibility effects were beginning to be recognized and accounted for in extensions of steady aerodynamics, first into multiple lifting line methods for oscillating wings, and later to lifting surface methods. A general lifting surface theory for finite wings based on Prandtl's acceleration potential (ref. 19) was given by Kussner (ref. 20) in 1940 in the form of an equation relating the unknown load distribution over the surface and the known velocity normal to the surface by means of a quantity known as Kussner's kernel function, K, (which represents the normal downwash induced at any point by a unit point load). However, it was not until 1954 that a general explicit expression for K was developed at the then NACA Langley Aeronautical Laboratory (ref. 21), and in 1966 a simple compact form of the kernel function for general nonplanar lifting surfaces was published (ref. 22). Two solution methods for the singular integral defining the kernel function which still are in wide use were developed over the period 1954-1975 - the kernel function method (ref. 23) and the doublet lattice method (ref. 24). Garrick and Reed remark in reference 16 that numerical lifting surface methods utilizing the velocity potential also are "in contention"; that discrete numerical methods initiated for steady incompressible flow by V. M. Falkner were extended to oscillating flows by W. P. Jones (ref. 25) and that more recent work with velocity potential by Morino and associates (refs. 26-28) has produced integral equation methods which can go beyond the linear approximation.

In the late forties and early fifties flutter at supersonic speeds began to be studied more seriously as speeds in dives could readily become supersonic. Several so-called "Mach box" methods of analysis were developed starting in 1955 (ref. 29, for example) and the kernel function approach was later extended at LaRC to supersonic speeds (ref. 30). Also, a modified strip analysis was developed at LaRC (refs. 31 and 32), suitable for subsonic, transonic, supersonic, and hypersonic flutter calculations, which still is in use today (refs. 33 and 34).

It was in the transonic speed range that there was a notable lack of progress in unsteady aerodynamics analytical methods development. However, the advent of flight at transonic speeds brought a host of new and challenging aeroelastic problems which emphasized the need to address this part of the flight regime. In the mid-to-late seventies the Loads Division (later the Structures and Dynamics Division and the Loads and Aeroelasticity Division) supported and contributed to several studies aimed at addressing this problem. Reference 35 summarizes many of these and some other studies. The problem area also was being attacked at the NASA Ames Research Center by in-house and contracted efforts (see, for example, references 36 and 37). These studies formed the basis for later concentrated efforts to address the transonic flow regime and other nonlinear considerations to be discussed later. It should, perhaps, be mentioned that except for references 36 and 37 the unsteady aerodynamic analytical methods mentioned so far are based on classical linear theory and fall at the very bottom of the hierarchy of governing equations as shown in Table 1. This table shows the governing equations, the associated assumptions, the applicable flow regime and aeroelastic applications, and some example "codes", from the relatively simple at the bottom to the most sophisticated at the top.

While significant advances had been made in aeroelastic analysis through the early fifties, researchers and designers still wanted and needed complementary experimental data. As transonic speeds were being realized and no accurate transonic unsteady analysis capabability existed, the need for a suitable wind tunnel for determining the aeroelastic and flutter characteristics of new high-speed aircraft became apparent to A.A. Regier at the Langley Aeronautical Laboratory who proposed in 1951 that the NACA construct a large transonic wind tunnel dedicated to research and tests in the field of aeroelasticity. The proposed facility was to be as large as feasible to enable accurate simulation of airplane details; be capable of operating over a wide density range in order to simulate various altitudes; use freon gas as the test medium to enable the use of heavier, less expensive models, permit higher Reynolds numbers and lower model vibration frequencies, and require less power; and be capable of operating at Mach numbers up to 1.2. As usual, the proposal required some selling and it was not until 1955 that the proposal was implemented by starting conversion of the Langley 19-Ft Pressure Tunnel to the 16-Ft Transonic Dynamics Tunnel. The tunnel is fully described in reference 38. The TDT became operational in 1960 and has since been used almost exclusively to support research and development testing in the field of aeroelasticity. The operational capabilities of the tunnel have been changed little over the years except that the maximum density capability has recently been increased by fifty-percent at Mach numbers greater than 0.7. Figure 1 depicts the TDT and some of the important elements of aeroelasticity it supports. Figure 2 shows some of the many high performance aircraft that have undergone flutter studies in the tunnel. Reference 18 discusses some of the more significant aeroelastic research and testing in the facility since it became operational to 1981.

Some of the major contributions by the tunnel staff to furthering the state of the art of predicting and controlling aeroelastic characteristics include perfection of a cable mount system so that models essentially could be free-flown in the tunnel (with greater than 1-g lift capability if needed), development of gust load simulation capability in the tunnel, various innovative methods for mounting models on the sting or sidewall in such a way that various "rigid body" degrees of freedom could be simulated, development of "subcritical"flutter testing techniques, and development of methods for predicting static and dynamnic loads on vertical standing launch vehicles due to ground winds. Regarding the last contribution it is interesting to note that the new TDT became operational in a period when emphasis was shifting from "aeronautics" to "space" and there were those who worried that the tunnel had arrived on the scene too late. However the TDT always was fully occupied and nearly half the research and development work in the tunnel in the sixties and early seventies was concerned with aeroelastic characteristics of launch and space vehicles including elastic stability and buffet loads on the Mercury-Atlas, Titan, Gemini, the Saturn series of launch vehicle, and the Space Shuttle, and ground wind loads studies of all the major U.S. space launch vehicles. Figure 3 shows some of the shuttle models studied in the TDT.

The "Staff"

Soon after the dawn of the "age of powered flight" the flying machines were designed by an individual (and usually test flown also by the designer). Over the years however the design of an airplane necessarily became a team effort with specialists in structures, aerodynamics, etc. all contributing. Similarly, in the field of aeroelasticity initially a few individuals were making very significant contributions to the understanding and treatment of this "problem" area - men such a T. Theodorsen, I. E. Garrick, and H. G. Kussner. However, as aircraft designs became more sophisticated and flight operational boundaries expanded, the scope of aeroelastic effects also grew and advances, particularly in analytical development, were made in more measured steps by many building on the previous work of others. The "flutter guy" in the major aircraft companies became the "flutter group," aeroelasticity research in the universities accelerated, and the few "theoreticians" in unsteady aerodynamics at the then Langley Aeronautical Laboratory grew to "section size" by the early fifties.

Although it has not been possible to determine an exact number from actual records, discussions with "old timers" at Langley indicate that from the early fifties to the late sixties the number of personnel working in the aeroelasticity area at Langley gradually rose to about forty. Although some were scattered through several organizations, most were in the Dynamic Loads Division. In a major reorganization at Langley in 1970 the Dynamic Loads Division became the Loads Division with 39 professional researchers working in aeroelasticity (including gust loads work but not including "vehicle vibrations" or "aerothermoelasticity" work). By 1978, due to attrition and another reorganization (the Loads Division became the Structures and Dynamics Division) the number of "aeroelasticians" had dropped to 24. Another reorganization in 1980 "cut the pie" a different way to create the Loads and Aeroelasticity Division (LAD) and the number working aeroelasticity increased to 34 by 1983. The trend was a welcome reversal because it was becoming increasingly apparent in the late seventies and early eighties that the trend towards increasingly flexible, high performance, unique designs that attempt to employ aeroelastic effects to good advantage was accelerating and creating bountiful opportunities and challenges in aeroelasticity. However, some of this staff is working the "new" area of "active controls" so that the number working in "traditional aeroelasticity" is considerably less than the staff of 39 in 1970.

At a strategic planning exercise in the summer of 1985 that addressed the personnel resources needed to meet planned technical goals a management goal was set to maintain the aeroelasticity staff at least at the present level. Since then, however, the number of the professional staff in the LAD working aeroelasticity has dropped to 32 in December, 1985. Also, the retirement of more LAD aeroelasticians is imminent within the next year and the future hiring authority appears to be extremely limited. Maintaining even the current level of staffing will thus be a worthwhile but challenging endeavor.

Selected studies of the LaRC Loads and Aeroelasticity Division and others during this period now will be presented with attention paid to unresolved questions. Then, based on the results of these studies and on perceptions of design trends, future research needs in aeroelasticity will be discussed.

RECENT ACCOMPLISHMENTS

Recent accomplishments (sometimes with limitations) are synopsized in this section from more detailed summaries given in Appendices A, B, and C.

The accomplishments are divided into three primary areas: Unsteady Aerodynamics, Experimental Prediction of Aeroelastic Characteristics, and Active Control of Aeroelastic Response. Unsteady aerodynamics is broken down further into Finite Difference Codes, Integral Equation Methods, and Unsteady and Steady Pressure Measurements.

Unsteady Aerodynamics

The past several years have seen steady progress being made in our ability to predict both steady and unsteady airloads needed for calculation of aeroelastic characteristics such as flutter, divergence, and static lifting surface deformations. The quantum leaps in computer capability since the sixties has permitted aerodynamic theory and code development that now is making significant progress up the hierarchy of flow equations. This hierarchy (figure 4) is topped at the highest level with the Navier-Stokes equations which contain terms accounting for the characteristics of all types of fluid flow one might encounter in the "real world"; i.e. viscosity, vorticity, compressibility, gravity effects, etc. The complete solution of these complex equations has not been practical from considerations of both computer capability and the cost of analysis. Therefore simplifying assumptions bring us down the hierarchy ladder to the Euler equations (viscosity neglected), to the full potential equations (viscosity, vorticity neglected), to the transonic small disturbance potential equations (limited to small thickness, incidence and amplitudes of motion, attached flow, and weak shocks), to the linear potential equations (essentially all the above limitations plus the limitation to Mach numbers outside the transonic range).

Historically the equations have been solved and applied at the lower levels with attempts to progress up the ladder as the needs and computational capabilities developed. Basically, two approaches to the solution of the unsteady potential flow equations have evolved. The "integral equation" method expresses the unknown flow velocity or potential in terms of a surface integral containing the linear terms of the governing partial differential equation plus a volume integral containing the nonlinear terms. For solution, these integrals are discretized by surface panels and finite-volume elements respectively. The "finite difference" and "finite element" methods, on the other hand solve the partial differential equation directly throughout a gridded flow field. Although the finite element approach recently has been used with some success in addressing steady aerodynamics and aero-thermo-structural problems in bounded domains, difficulties are being encountered in extending the approach to unsteady aerodynamics in the unbounded domain. Both the integral equation and finite difference approaches have their advantages/disadvantages and proponents/opponents. In application, although both approaches involve "computation," finite difference has become the more widely used approach in the "computational fluid dynamics" (CFD) area. With the finite difference approach, it has been customary to consider "two-dimensional" flow cases first before progressing to the more difficult and practical "three-dimensional" capability. With the integral-equation formulation that step is not necessary. A measure of advancement is the capability to handle wing alone, two surfaces, or complete configurations. PANAIR (ref. 39) and SOUSSA (ref. 40) and SUSAN (ref. 41) are examples of early integral-equation codes developed under NASA contracts and grants, and XTRAN2L (ref. 42) and XTRAN3S (ref. 43) are examples of the latest finite difference approaches developed with Air Force and NASA resources. Once the algorithms are developed and the codes written there is a period (some would say unlimited period) of "applications" in which shortcomings of the original algorithm or code are identified both by comparing results with other analyses and with experimental results, coding improvements are made to increase efficiency and accuracy, and the ranges of applicability expanded. This critical period may be typified by some examples of recent improvements in the XTRAN2L and XTRAN3S codes at LaRC, some accomplishments in the "integral equations" area, and some examples of recent accomplishments in unsteady pressure measurements. These are discussed in Appendix A, and are briefly summarized in the following sections.

Finite Difference Codes. - As discussed in Appendix A, most of the accomplishments in this area were focused on improving the efficiency and extending the range of applicability of the so-called XTRAN small perturbation finite difference codes. These included extending the frequency, Mach number, and angle of attack ranges and the accuracy and efficiency of the two-dimensional XTRAN2L code. The techniques involved in these improvements are being incorporated into the three-dimensional code XTRAN3S. Also, the manner in which the boundaries of the computational domain are considered have been changed to eliminate inaccuracies caused by reflected disturbances from the boundaries. Work in this area is continuing to minimize the number of required grid points to make the program more efficient. Both XTRAN2L and XTRAN3S have been evaluated on several different configurations. The correlation of the calculated pressure with experimental data ranged from good to poor. It is believed that the incorporation of more adequate computational grids and viscous modeling will improve the predictions. Also, further development is needed to treat thick supercritical wings with shockwaves, and to increase the numerical stability to reduce the resources needed for application of these codes.

In another improvement on the basic XTRAN2L code which was developed for isolated planar wings, the time-marching solution procedure of the code has been extended to admit an additional lifting surface which allows the calculation of unsteady transonic flow fields about two-dimensional interfering airfoils. This capability recently also has been incorporated into the three-dimensional transonic code XTRAN3S. The next step is to develop a wing-fuselage capability. Other near term additional capabilities needed and being worked for XTRAN3S include boundary layer modeling for viscous calculations and modeling the non-isentropic strong shock condition.

In addition to these improvements in the two codes, certain deficiencies that exist in classical transonic small disturbance (TSD) potential theory on which the codes are based (non-unique solutions for certain flow conditions, for example) have been corrected by means of several modifications to the TSD equations. These modifications have been implemented into XTRAN2L and will be extended to XTRAN3S. Also, since the TSD potential equations on which the XTRAN codes are based do not consider viscosity, and to improve the accuracy, an existing quasi-steady integral boundary layer method has been coupled with the XTRAN2L code, and will be incorporated also in the XTRAN3S code. Work is starting at the Unsteady Aerodynamics Branch at LaRC on development of a full potential equation finite difference method for aeroelastic analysis. This advanced method must treat complete vehicles (wing/body/canard, tail/vertical tail) in the subsonic/transonic/supersonic speed ranges involving both vortex flows and flows with strong viscous interactions. Ideas for 3-dimensional methods are first being tested in two-dimensions. These include body-conforming, time-dependent grid generation methods, corrections for strong shock anomalies, and inclusion of viscous boundary layer and vortex flow models.

Integral Equations Methods. - A less extensive effort has been underway for some time at LaRC to implement an "integral equation" computational approach mentioned earlier. Application of a generalized Green's function method to the full, time-dependent potential-flow equation leads to an integral equation for the velocity potential at any point in the flow, including points on the surface of a body in the flow. The SOUSSA (Steady, Oscillatory, and Unsteady Subsonic and Supersonic Aerodynamics) P1.1 program is a panel-method code which implements this integral equation for linearized subsonic flow in the complex-frequency domain, and is applicable to general shapes such as complete aircraft having arbitrary shapes, motions and deformations. Subsequent to the completion of the initial form of the SOUSSA P1.1 code several significant improvements have been incorporated to increasethe efficiency and accuracy of the code, and others are known to be needed (ref. 44). In addition to the subsonic capability of the SOUSSA program a supersonic proof-of-concept surface panel code has been written to implement linear theory algorithms developed under NASA grants (refs. 45 and 46). Like SOUSSA, this program also is applicable to vehicles having arbitrary shapes, motions, and deformations. Validation and application of the code have begun.

For flow conditions that introduce nonlinearities in the equations (e.g. high subsonic and transonic flows) the volume integral of the nonlinear terms is retained and integrated numerically, and the associated transonic-small-perturbation computer program is called SUSAN (Steady and Unsteady Subsonic Aerodynamics - Nonlinear) (ref. 41). The proof-of-concept SUSAN code has been limited for convenience and economy during the development to rectangular planforms and low frequencies. Both limitations, however, can be removed. Additional nonlinear terms could be readily included in the volume integral to solve the full potential equation. This activity has been superseded, however, by the recent development of a full-potential integral-equation method for subsonic or transonic flow past bodies of arbitrary shape at low-to-high angles of attack, including vortex-type flow separation and shock waves (ref. 47). Efforts are underway to incorporate major but straightforward improvements in the demonstration code to achieve the full generality of this method. The modifications will include, for example, taking account of entropy changes across shock waves in order to extend the usefulness of the potential-flow method into ranges of flow variables where Euler solutions would otherwise be needed.

Finally, a scalar/vector-potential (SVP) formulation, equivalent to Cauchy's equation of motion or the Navier-Stokes equations, has been developed for general three-dimensional, unsteady, compressible, viscous, heat-conducting flow (ref. 48). This formulation employs the classical Helmholtz decomposition of any vector field into the sum of an irrotational and a solenoidal field. An integral-equation implementation is an appropriate method of solution and may offer possibilities for significant improvements in computational efficiency relative to direct solution of the Navier-Stokes equations in primitive variables. A proof-of-concept code for two-dimensional flow is being written.

These developments in finite-difference and integral-equation methods illustrate the manner in which the capability and efficiency of state-of-the-art programs are being improved and how the needs for treatment of more complicated flow problems are being addressed. For example, the need to model blended-body complete configurations with the attendant blunt bodies in the transonic speed range characteristic of advanced fighter and "low observable" concepts call for algorithm and code development at the full-potential or higher level. The effects of vortex-type flow separations must be taken into account because of its typically detrimental effect on structural design loads and flutter. The influence of viscosity and flow separation must be considered in order to calculate accurately the forces generated by deflected and deflecting control surfaces as well as the forces on advanced fighter aircraft in combat maneuvers at high angle of attack and highly transient conditions. The activities described here must be pursued vigorously so that needed information, understanding, and technological capabilities will be available as soon as possible for use in the study and design of advanced aircraft, especially combat aircraft.

Unsteady and Steady Pressure Measurements. - In parallel with unsteady aero code development/evaluation there has been a progressive unsteady pressure measurement program to provide an expanded data base for a variety of configurations, to permit code verification, and to help better understand flow field/body interaction phenomena. The experimental program was started several years ago and has progressed through several aerodynamic configurations. Four large wing models have been tested in the LaRC Transonic Dynamics Tunnel (TDT) under the initial program (see Appendix A). The first model was a rigid highly swept, sharp leading edge, clipped delta planform configuration sidewall-mounted through a "splitter plate" and capable of being oscillated in pitch. Two control surfaces, one leading edge and one trailing edge, could be oscillated independently. The second pressure model was a rigid semispan supercritical wing representative of current energy efficient transport designs. The model has 10 oscillating control surfaces but only an inboard and an outboard trailing edge surface were used in the initial tests to generate unsteady airloads. The third unsteady pressure model tested in the TDT had an unswept, rectangular planform and a supercritical airfoil section. Unsteady pressures were generated by oscillating the wing in pitch at several frequencies. The fourth wing tested was similar in planform to the second wing but differed significantly in that the second wing model was made as rigid as possible whereas the fourth wing actually was a very flexible wing off a drone vehicle used in the flight research program, Drones for Aerodynamic and Structural Testing (DAST). Each of these models served to evaluate the ranges of applicability and accuracy of various analytical methods and to provide a better insight into the mechanism of unsteady transonic flow.

The initial unsteady pressure measurement program has been carried a step further with tests of a two-dimensional rigid supercritical airfoil model at Reynolds numbers up to 35 million in the LaRC 0.3m Transonic Cryogenic Tunnel (TCT). In addition to providing new data on Reynolds number effects on unsteady aerodynamics, the tests provided an opportunity to address some of the known and unknown problems associated with dynamic testing at cryogenic temperature and is considered a precurser to future desirable tests at high Reynolds numbers including an oscillating control surface test in the 0.3m TCT and a three-dimensional wing in the LaRC National Transonic Facility.

The results of the four pressure models tested in the TDT and the two-dimensional model tested in the 0.3m TCT are summarized in Appendix A.

Two other unsteady pressure studies in the Transonic Dynamics Tunnel (TDT) are planned for the near future. The first involves the measurement of unsteady pressures on a canard and a wing due to oscillating the canard. This study is needed to determine unsteady aerodynamic interference effects of this increasingly popular configuration for fighters and to validate computer codes being developed to handle such cases. In addition to surface pressure measurements a "laser vapor screen" technique will be used to visualize the interfering flow fields. This study is planned for 1986. In the 1986-1988 period a "benchmark" combination flutter/unsteady pressure model will be fabricated and tested in the TDT. This three-dimensional model will have a simple geometry and high quality mass, stiffness, and surface definition. Surface pressures and wing deformations, both static and dynamic, will be measured and the flow field will be determined from subcritical conditions all the way to hard flutter for validation of flutter analyses using the advanced unsteady aerodynamics codes. This study should not only validate the analyses, but in addition should help diagnose deficiencies in the aero codes.

Experimental Prediction of Aeroelastic Characteristics

Aeroelasticity problems encountered by high speed aircraft most often arise in the transonic speed range where aerodynamic theory is least developed, so heavy reliance is placed on wind tunnel models to aid in showing new designs to be safe from flutter and divergence and to have acceptable buffet characteristics, to evaluate solutions to aeroelasticity problems, and to study aeroelastic phenomena. Most of the experimental studies in aeroelasticity at the LaRC are conducted in the Transonic Dynamics Tunnel. Reference 49 presents a summary of many of these activities since 1981. Some recent studies that have both answered and raised questions regarding aeroelastic phenomena (reviewed more fully in Appendix B) now will be summarized briefly.

The effects of wing static deformation on flutter speed was studied experimentally and analytically. The experiment used a transport type supercritical semispan flutter model which exhibited a substantial degradation in flutter speed as wing-root angle of attack increased over a limited Mach number and angle of attack range. The trends of the experimental data were duplicated in flutter boundaries calculated by a two-dimensional transonic finite difference computer code in which static angles analogous to washout of angle of attack at the tip of an aerodynamically loaded wing were used as a function of Mach number and dynamic pressure. This indicated the angle of attack effects were due to static torsional deformations of the wing. Corresponding three-dimensional calculations by a modified strip analysis, which used as input experimentally determined (from a stiff aerodynamic model) spanwise distributions of section lift-curve slope and aerodynamic center showed a tendancy toward similar behavior, although the backward turn of the measured transonic dip was not reproduced. However, the analysis did indicate a double "transonic dip" in the flutter boundary previously seen in flutter tests of supercritical wings.

Angle of attack effects also have figured prominently in other aeroelastic studies conducted in the TDT, including limit cycle oscillations of the B-1 wing at certain angles of attack that were encountered in flight and essentially were duplicated in flutter model tests in the TDT.

In the pressure tests of the high aspect ratio DAST ARW-2 wing mentioned previously, another unusual flutter boundary not predicted by theory was encountered. The instability occurred at nearly constant Mach number from very low dynamic pressures up to the tunnel limit. Additional testing of the wing is planned to define further the mechanism of the instability, to determine the effectiveness of the active control system in controlling the instability and to extend the measured instability boundary to higher dynamic pressure making use of the increased operational capability recently provided to the TDT.

In another study in the TDT an unusual "body-freedom flutter," caused by adverse coupling of rigid-body-pitching and wing bending motions, was demonstrated on a model of a forward swept wing (FSW). Because this flutter mode was of concern for the X-29A aircraft a half-scale semi-span model of an early FSW aircraft design was tested on a cable and rod sidewall mount system that provided the necessary pitch and plunge degrees of freedom. The data were used to validate aeroelastic analysis of the vehicle.

An example of the need to be sensitive to the effects of aeroelastic stability of innovative aerodynamic configurations that bring increased aerodynamic efficiency is illustrated by a series of studies in the TDT of the effects of "winglets" on wing flutter. Tests of three different transport-type swept-tapered wing configurations showed that the flutter speed of the basic wing was reduced in varying degrees by the aerodynamic effects of added winglets.

In addition to these "research type" studies there have been several "flutter clearance" tests in the TDT. The most recent of these have been for the F-16 and F-16E fighter with various external store configurations.

Active Control of Aeroelastic Stability and Response

Figure 5 illustrates the functions that are usually associated with active control of aeroelastic stability and response. Because of its impact on safety of flight, flutter suppression is probably the concept furthest from practical implementation and has received significant attention. The concept is to increase the flutter speed of the vehicle through the use of active controls. The benefit to be derived from flutter suppression usually is reduced structural weight. Gust load alleviation and ride quality improvement apply to flight through atmospheric turbulence. Gust load alleviation allows reduced structural weight by reducing wing loads. Ride quality control improves the passenger ride comfort and reduces crew fatigue. Reduced static stability is an active control function that does not directly involve aeroelastic response although structural flexibility may have to be considered. The benefits are reduced drag, a smaller horizontal tail, and an increased useable c.g. range. Maneuver load control also is used to reduce wing loads and the benefit is reduced structural weight. Again, flexibility effects may be important.

Since the early seventies there has been a growing recognition of the potential gains in aerodynamic efficiency and structural weight savings that can be realized through the use of active controls to alleviate gust loads, improve ride quality, and reduce fatigue (thereby extending the useful life of the airplane); reduce maneuvering loads; and suppress airframe instabilities such as flutter and divergence. As a result, there have been significant advances on both the analytical and experimental fronts of this relatively new technology area. (An indication of the growing realization of the importance of servo-feedback control functions in aeroelasticity is the increasing use of the expanded term "aeroservoelasticity" to describe the technology area).

Basically, experiments (wind-tunnel and flight) have been used to validate theory or analysis, to evaluate feasibility, and to demonstrate predicted benefits. Reference 50 summarizes and attempts to put into perspective (relative to "rigid-body" and "aeroelastic" control functions) the results of the various wind-tunnel and flight experiments performed under the banner of "active controls" through 1983. It is worth mentioning that the first practical successful wind tunnel demonstration of the feasibility and benefits of active control technology for elastic mode control (for the control function most hazardous and difficult to achieve-flutter suppression) was accomplished in the LaRC TDT, using a cropped-tip delta simplified model of a supersonic transport wing (reference 51). Other pioneering studies included wind tunnel and flight demonstrations of the effectiveness of flutter mode control and ride quality control using a modified model and full scale B-52 "Control Configured Vehicle," (ref. 52 and 53 for example), wind tunnel and flight studies of an Active Lift Distribution Control System on the C-5A to reduce the incremental inboard-wing stresses experienced during gusts and flight maneuvers (refs. 54 and 55 for example), and wind tunnel studies of active control law parameter sensitivities for flutter suppression on a DC-10 aeroelastic model wing (ref. 56). Other accomplishments (reviewed in Appendix C) will be summarized here briefly.

A modified flutter model of the F-16 was used in the TDT to demonstrate that active controls could be used to negate the decrease in both the symmetric and antisymmetric flutter speeds often brought on by the carriage of external stores. Also a flutter model of the YF-17 has been used in a series of tests in the TDT to demonstrate the feasibility of progressing to digital, adaptive active control for flutter suppression.

In addition to wind tunnel studies, the unique flight research program DAST provided a focus for evaluation and improvement of synthesis and analysis procedures for design of active control systems on wings with significant aeroelastic effects. Much was learned of the practical design and synthesis of flutter suppression systems and integration of other control functions with these systems before the program was terminated due to severe funding restrictions.

In addition to the experimental work described, a significant amount of work has been performed to develop analysis and synthesis methods required for the efficient and effective design of integrated active control systems, which, in turn, must be an integral part of the aircraft design process. This complex task requires the use of efficient multidiscipline computer programs. Work at the LaRC has provided a solution to the problem of incorporating unsteady aerodynamics into such programs. The primary emphasis of the work in the Loads and Aeroelasticity Division has been the development of such design methodology, with emphasis on the use of optimal control theory. A recent significant achievement has been the development of a new approach that allows the designing of a low-order control law from optimal control theory while maintaining optimality in the order reduction process.

NEW CHALLENGES

Unsteady Aerodynamics

The accomplishments cited earlier and others depicted in the recent literature indicate a steady advancement in the capability to predict both steady and unsteady aerodynamic loads under increasingly realistic conditions but it also is evident that there still are many unknowns and limitations in the applicability of present codes that need to be explored, and that the continued march up the hierarchy of flow equations will be challenging indeed. The principal objective of future work continues to be development of a capability for calculating the steady and unsteady viscous flows around complete aircraft with complex configurations, including blended components, engine nacelles, external stores, etc. for both attached and separating flows throughout the subsonic, transonic, and supersonic flight regimes, and the application of the aerodynamic predictive capability to accurate aeroelastic analysis methodology.

The state-of-the-art transonic small disturbance code XTRAN3S will be enhanced to handle (in addition to multiple lifting surfaces) fuselage, pylon/nacelle/store, tip missile, and winglet and control surface aerodynamics.

An advanced code based on the full potential equation with better geometry definition, boundary conditions on actual surface position, and with more efficient, robust, and affordable modern algorithms is needed, and initial development has begun.

A viscous boundary layer capability (could be incorporated in, or used in conjunction with the above codes) is needed that can handle at least attached flow and mild separation cases and is useable at speeds from subsonic to low supersonic. Later capabilities should handle fully separated flow at higher Mach numbers.

There is also much that could be done to expand the usefulness of integral equation-methods. The subsonic UTSA surface-panel program needs to be completed as a stand-alone program as well as to generate the velocity field induced by body-surface singularities in transonic flow and/or flow with vortex-type separation. The supersonic proof-of-concept code needs to be validated and the capabilities incorporated as supersonic modules in UTSA. A number of major improvements in the transonic/free-vortex code (ref. 47) have been defined and need to be incorporated into the code. For example, the combinations with UTSA mentioned above will permit calculations for aircraft of arbitrary shape. Completion of these activities would provide efficient and unified treatment of flow over vehicles having arbitrary shapes, motions, and deformations at subsonic, transonic, and supersonic speeds at low tomoderate angles of attack including leading edge vortex flows. Table 2 summarizes the integral-equation activities and indicates the unity of the program. As indicated previously, the UTSA surface-panel program for attached flow may contain both subsonic and supersonic modules in a single program. If the volume integral module is included with UTSA, the result is a program implementing the full-potential equation for transonic nonlinear attached flow. If the hybrid-vortex module (ref. 57) representing the surface integral for free vortex sheets is also added, the resulting code could treat transonic flow with vortex-type separation.

Finally, the scalar/vector-potential (SVP) formulation should be developed and assessed as a potentially more efficient method for calculating viscous flows than solution of Navier-Stokes equations in primitive variables. Such capabilities are, of course, needed for high angles of attack where flow separation from surfaces may occur but are also needed even for low angles of attack when control-surface deflections are large enough or shock waves are strong enough to cause significant boundary-layer thickening or separation.

A significant continuing challenge will be the appropriate division of limited resources between the "integral equations" and "finite difference" approaches. A "high risk" challenge would be the pursuit of the application of the finite element approach to unsteady aerodynamics.

The above activities are seen to be "main-line" development of finite difference codes geared to be affordable (but expensive) for aeroelastic analyses for final design. More exotic (and much more expensive) methods are in the offing. Euler equation-based codes are in development and in use for steady flow but a concensus has not yet developed on what the present versions can or cannot provide. Once the steady flow situation is clarified, the difficult extension to the unsteady flow regime may be undertaken. Time-accurate Euler equation codes may be required to analyze vortex-driven aeroelastic effects such as those that have been postulated as being experienced on the B-1 aircraft. An important current difficulty with Euler equation codes for leading edge vortex flow calculations is the inability to predict the location of the separation line for blunt leading edges. The free-vortex method for transonic flow (ref. 47) combined with the hybrid-vortex method (ref. 57) and the improved surface panel method discussed above should provide a powerful tool for calculating transonic and/or free-vortex flows around arbitrary aircraft configurations with sharp leading edges or with assumed separation line locations. Establishing the separation line on a vibrating wing however, is a tough viscous flow problem, but may be amenable to treatment by the scalar-vector potential method. Much activity is needed in this area since vorticity (rotational flow) capability will become increasingly important for supermaneuvering fighter concepts, and already is needed for known load-induced wing store flutter problems on the F-14 to F-18 series aircraft.

Experimental Prediction

The number of new, unique (some might even say bizzare) geometric configurations being proposed for increased aerodynamic and structural efficiency and for newly identified operational requirements (both military and civil) seem to be increasing exponentially. Some recent examples, some of which are shown in figure 6, are the forward swept wing (actually not a new aerodynamic concept, but newly accepted due to new enabling structural technology), oblique wings, X-wings, twin fuselage transports (re-visited), super high aspect ratio lightweight long-endurance high altitude observation configurations, low-observable "blended" configurations, highly maneuverable "agile" fighter configurations, etc. Earlier innovative aerodynamic concepts such as the supercritical airfoil, winglets, and "speed pods" have brought with their aerodynamic performance benefits added aeroelasticity problems. One of the great challenges will be to determine in advance what, if any, are the aeroelastic problems associated with the new concepts. "Trend" studies using simple models of these advanced concepts may give some clues and should be done selectively but with the understanding that some aeroelastic characteristics may be overlooked until precise (expensive) dynamically scaled aeroelastic models of a specific configuration are tested.

An imminent conflict that will have to be resolved is the division of limited available test time in the TDT and other tunnels between "research" type models to gain an understanding of new or potential aeroelastic phenomena or tests to validate new analytical methods, and "flutter clearance" tests of actual new configurations. An additional simple, small-scale, open-circuit tunnel with a minimum of frills could help materially to alleviate this dilemma.

Also, to help understand better the flow fields involved in aeroelastic phenomena and to better evaluate the new sophisticated analytical methods under development, a capability to "visualize" the flow field at least qualitatively, including shock/boundary layer interaction is needed as is a means for measuring model static and dynamic deflections.

As the new aircraft designs become more sophisticated so too the aeroelastic models become more complicated, expensive, and fragile which makes even more important the development of "subcritical" testing techniques, more efficient "on-line" data analysis, and less inhibiting or interfering model mounting methods.

A challenge in the aeroelastic prediction area that has not been met adequately in recent years and which again is becoming increasingly important due to new configuration concepts and operational requirements is that of predicting the intensity of buffet of stability and control components such as vertical and horizontal tails. Although some progress is being made in the calculation of separated flow fields on airfoils (ref. 58 for example) little has been done since the mid-to-late seventies to advance the state of the art for predicting the buffet response of aircraft components (ref. 59) other than some refinement of then existing techniques (refs. 60, 61 for example). The last buffet study per se conducted in the TDT was in 1972 (ref. 62). This sparcity of documented activity arose from the perception that buffet was not a structural problem for the very strong and stiff high-load-factor fighters then being developed (little "urgency") and the configuration-dependent nature of buffet response. The twin tail fighters currently are suffering because of this neglect (ref. 63, for example).

The configuration dependency of the problem compounds the difficulty of early assessment of potential buffet problems before exact geometry and structural properties of a proposed concept are known. Increased sensitivity to the potential problem and the increasing emphasis on combat maneuvers at very high angle of attack, however, may provoke an interest in incorporating into flutter clearance models the capability to assess the likely buffet intensity as the buffet boundary is penetrated. The capability is achieved primarily, of course, by adding strength to the flutter model to permit high angle of attack testing at reasonable levels of dynamic pressure (Reynolds number). This, in itself, can be a challenging task.

Active Control of Aeroelastic Stability and Response

The advances that have occurred in analytical methods are allowing certain non-flight-critical active control functions to be considered during aircraft preliminary design. However, considerable work remains to be done before math models can be used with the necessary confidence for flight critical active control functions. All of the analytical methods mentioned in the section on accomplishments were based on linear techniques (both the equations of motion and the aerodynamic equations). Since the critical speed range for flutter often is transonic and the use of supercritical airfoils is increasing, the interfacing of new nonlinear transonic unsteady aerodynamic force computations with nonlinear structural dynamics and discrete, nonlinear control system equations will become increasingly important (and challenging). The interfaces will be needed for both analysis and control law synthesis. In the control system design area, future emphasis will be on functionally integrated control system design for total mission control, including "adaptive" concepts, all oriented to implementation by digital computer.

The achievements that have been made in the field of aeroelasticity are impressive, but so are the outstanding needs. We have been chipping away at the "frontier" but the rate of progress is inadequate. Already flight projects are underway (the oblique wing and X-wing, for example) which find our level of aeroelastic expertise inadequate. Aeroelasticity design tools with uncomfortably little validation are therefore necessarily being used. It is likely that the more demanding design of the advanced tactical fighter, intended to enter service in the mid-nineties, also may have to proceed with deficient understanding in the aeroelasticity area unless more committment is made to increasing the rate of progress.

CONCLUDING REMARKS

Aeroelasticity has evolved from a "problem avoidance" or "fixing" technology to one which can provide substantial performance improvements and in some cases even may be an "enabling" technology. Aeroelasticians are beginning to wear white hats! The increased importance of aeroelasticity in the design process has been recognized in the industry by increasing staffs and in the universities by expanding curricula. The staff working aeroelasticity problems at the LaRC peaked in the early seventies and decreased substantially in the late seventies after the "moon race" was won, and NASA's authorized complement was severely reduced. The current staff is substantially below the peak level. In view of an expected increase in the near-term retirement rate and very limited hiring authority, maintenance of even the present level of staffing will be difficult but essential.

The recent significant accomplishments in unsteady aerodynamics, both analytical and experimental, the experimental prediction of aeroelastic characteristics, and the active control of aeroelastic stability and response that have been discussed are noteworthy as advancements in the state of the art, but the activities also in many instances have indicated the need for more in-depth research in the specific areas and a general broadening of the scope to address a host of new aerodynamic and structural concepts for performance improvement. Although there have been significant accomplishments in aeroelasticity at the NASA LaRC along a broad front, not only is the front not advancing rapidly enough to be ready for future advanced design concepts, it is not even keeping abreast of already proposed flight programs/projects.

Some of the identified needed activities have been discussed in the section "New Challenges", but the real challenge for the LaRC is how best to allocate the limited available financial and manpower resources among the perceived opportunities and needs in aeroelasticity, and how best to get the most "bang for the buck" through cooperative efforts with the aerospace industry and academia similar to those being developed with the Air Force Flight Dynamics Laboratory.

APPENDIX A

RECENT ACCOMPLISHMENTS IN UNSTEADY AERODYNAMICS AT THE NASA LANGLEY RESEARCH CENTER

Finite Difference Codes

The program XTRAN2L is a derivative of LTRAN2, a two-dimensional finite difference code developed at the NASA Ames Research Center (ref. 37). This program was limited to low reduced frequencies and was sensitive to flow conditions. Modifications (see fig. 7) at LaRC included the addition of an unsteady term in the complete differential equation into the code, allowing accurate calculations for all frequencies (ref. 42). Also a monotone differencing scheme in the transonic algorithm is incorporated which considerably extended the Mach number and angle of attack range of the program. Nonreflecting boundary conditions were added which allowed a reduction of the extent of the grid and thus reduced computer costs. In addition a new finite difference grid was developed that considerably enhanced the accuracy of the results by eliminating spurious oscillations in the unsteady loads (ref. 64).

A key factor in developing and assessing the improvements was the implementation of a pulse-transfer function technique based on fast Fourier transforms to obtain unsteady airload frequency response functions from a single transient calculation. This provides airloads for all frequencies of interest with a significant computational savings. Comparisons with exact linear theory results for a flat plate airfoil permits rapid assessment of accuracy of the parameters, such as the grid, being investigated. Figure 7 gives a sketch of the influence of the grid and nonreflecting boundary conditions. The improved code XTRAN2L is more robust, more accurate, and reduces the computer costs by 33%. These techniques and improvements are being implemented in the three-dimensional transonic code XTRAN3S and in a full potential code that is under development.

The XTRAN3S transonic aerodynamics and aeroelastic analysis code, developed by Boeing Military Airplane Company under a USAF contract can handle modal aeroelastic analysis of isolated planar wings. The original version of the XTRAN3S computer program for three-dimensional transonic small perturbation unsteady aerodynamic analysis incorporated steady state conditions at the boundaries of the computational domain. This caused disturbances incident on the boundaries to be reflected back into the computational region (figure 7). The reflected disturbances can cause errors in the calculated unsteady aerodynamic loading. In a recent research effort, nonreflecting (characteristic) far-field boundary conditions, which simulate outgoing waves at the boundaries, were developed. These conditions absorb most of the disturbances incident on the boundaries. The boundary conditions were implemented in XTRAN3S and tested by calculating the unsteady forces on a flat plate rectangular wing with a pulse in root angle of attack α. The lift response C_ℓ and frequency response function for the unsteady lift curve slope C_{ℓ_α} were calculated with and without the nonreflecting boundary conditions. The calculations were made for a freestream Mach number, M, of 0.85. The lift response, which should return smoothly to zero after an initial transient, is at the left of figure 8. Use of the steady state far-field conditions results in oscillations in calculated lift well after the initial transient. When the nonreflecting conditions are used, the lift returns smoothly to zero. In the frequency response function, shown at the right of the figure, the XTRAN3S results are compared with those obtained with an exact kernel function method. Using the original far-field conditions causes spurious oscillations in the frequency response. When the new conditions are used, the oscillations are eliminated and good agreement with the kernel function method is obtained. Future plans include systematic studies to determine how close the far-field boundaries may be placed with no loss in accuracy of the computed results. This could lead to fewer required grid points and thus a more efficient computational method.

XTRAN2L and XTRAN3S already have been applied by the LaRC staff to several different configurations including four of the AGARD two-dimensional aeroelastic configurations and one of the three-dimensional configurations (ref. 65) and a low aspect ratio tailplane type configuration (ref. 66). The correlation of the calculated pressures with experiment ranged from good to poor. The authors of reference 65 anticipate that the incorporation of more adequate computational grids and viscous modeling, will significantly improve the predictions. Reference 66 concludes that further development is needed to treat thick supercritical wings with shock waves and that further effort is needed to increase the numerical stability (to permit large time steps) to reduce the resources needed for application of these codes. In addition to these evaluations, XTRAN3S is being applied, through cooperative efforts with the aircraft airframe industry, to the AV-8B, F-5, C-17, and an advanced transport wing to gain experience with and to further evaluate the code.

In order to correct certain deficiencies that exist in classical transonic small disturbance (TSD) potential theory on which XTRAN2L and XTRAN3S are based (non-unique solutions for certain flow conditions, for example) several modifications have been made to TSD theory and implemented into XTRAN2L (ref. 67). Modifications to the TSD equation are derived from a formal asymptotic development of the Euler equations including the effects of shock generated entropy. To first order, this results in the classical TSD equation while the second order terms provide the above noted modifications. This second order theory involves; a) a new streamwise flux formulation satisfying the exact Prandtl relation for shock jump conditions, b) the inclusion of shock generated entropy in the pressure coefficient evaluation and c) the convection of that entropy in the wake boundary condition.

To assess the impact of these modifications, steady and unsteady calculations were made for the NACA0012 airfoil using both the modified and unmodified small disturbance codes. The calculations indicate that the modifications resolve the nonuniqueness problem in static and dynamic cases, with the solutions giving good agreement with Euler code results. The modified flux and pressure coefficient evaluation place the shock in the correct position (within one grid cell of the Euler solution) with the correct strength, in terms of velocity (Prandtl relation is captured by the numerical scheme) and pressure. The convection of entropy in the wake constrains the shock motion to realistic locations (agreeing well with Euler results) for lifting flow cases while consistently modeling the entropy jump in the flow.

Figure 9 clearly shows the effects of the modified theory for the steady pressure distribution for a case in the middle of the nonuniqueness region. Similar agreement with the Euler code is obtained for cases outside the non-uniqueness region for the NACA0012 airfoil, and for NACA64A010 and NLR7301 airfoil cases.

The modified theory requires only minor coding changes in existing small disturbance algorithms and entails little increase in computational costs. The modified second order theory will be extended to the XTRAN3S three dimensional code to enable unsteady aerodynamic calculations on wings in strong transonic flows. Also, it is anticipated that the second order theory, when coupled with a viscous boundary layer capability, will improve agreement with experimental unsteady pressure distributions.

Since the transonic small disturbance potential equations on which XTRAN2L is based do not consider viscosity and in order to improve the accuracy, an existing quasi-steady integral boundary layer method has been coupled with the unsteady, inviscid, transonic computer code XTRAN2L (ref. 68). The coupling procedure is derived from the LTRAN2 viscous code developed by Donald Rizzetta for Ames Research Center. Several modifications to Rizzetta's procedure have been made, the most important being the ability to iterate the viscous-inviscid solutions at each time step and the inclusion of an explicit coupling technique which greatly accelerates convergence. To assess these modifications, viscous calculations have been made for several airfoils under both steady and unsteady flow conditions. The unsteady conditions investigated include harmonic oscillations as well as transient pulses for obtaining unsteady airload frequency response functions from a single transient calculation. Figure 10 shows results which have been obtained for an NACA 64A010 airfoil. The overall good agreement shown between the viscous calculations and experimental results is typical for the airfoils studied. These techniques will be incorporated into the unsteady 3-D transonic code XTRAN3S.

In another improvement on the basic XTRAN2L code which was developed for isolated planar wings, the alternating-direction implicit (ADI) time-marching solution procedure of the code has been extended to admit an additional lifting surface (ref. 69). The program now is capable of computing unsteady transonic flow fields about two-dimensional interfering airfoil configurations. To demonstrate this XTRAN2L multiple surface capability, selected steady results are presented for a closely-coupled configuration as shown in figure 11(a). In this example, the leading airfoil was placed one chordlength upstream of the trailing airfoil (measured from midchord to midchord in units of trailing airfoil chordlength) and one-quarter chordlength above. Mean angles of attack for both airfoils were $\alpha = 1°$ and the freestream Mach number was $M = 0.5$. The lifting pressure coefficient, ΔC_p, is plotted for both isolated and interfering configurations. In the upper-right of the figure, comparisons of lifting pressures calculated with the XTRAN2L code and with an independent vortex lattice code are given for the case of flat plate airfoils. The good agreement shown verifies the XTRAN2L code modifications. The distance between the isolated ΔC_p curves and the interfering ΔC_p curves represents the aerodynamic interference between the two lifting surfaces. For the configuration shown, the leading airfoil produces a downwash on the trailing airfoil thus decreasing its ΔC_p and lift. Conversely, the trailing airfoil induces an upwash on the leading airfoil which increases its ΔC_p and lift. In the lower-right of the figure, XTRAN2L pressure distributions for NACA 0010 airfoils at M = 0.5 further demonstrate the aerodynamic coupling between the airfoils. Mach number contour lines for this case clearly illustrate the strong interference effects.

This work was a preliminary effort to assess the incorporation of the multiple lifting surface capability into the three-dimensional transonic code XTRAN3S which currently is underway (ref. 70). Grid generation procedures for multiple surface configurations were developed by extending the existing procedures for single lifting surfaces. A sample planform finite-difference grid near a canard-wing configuration and transonic steady pressure distributions on the canard and wing upper surfaces are shown in figure 11(b). The multiple surface computational capability will permit the assessment of interference effects on transonic unsteady airloads and flutter. The next step is the development of wing-fuselage capability.

In addition to these LaRC improvements to XTRAN2L and XTRAN3S other improvements have been or are being implemented at Ames such as wing-tip launcher and missile capability and at Boeing (Pylon/Nacelle/Store modeling). At LaRC some near term additional capabilities being worked for XTRAN3S include boundary layer modeling for viscous calculations and modeling the non-isentropic strong shock condition.

Integral Equations Methods. - A less extensive, effort has been underway for some time at LaRC to implement an "integral equation" computational approach mentioned earlier. Application of a generalized Green's function method to the full, time-dependent potential-flow equation leads to an integral equation for the velocity potential at any point in the flow, including points on the surface of a body in the flow (Refs. 26, 40, 45). The SOUSSA (Steady, Oscillatory, and Unsteady Subsonic and Supersonic Aerodynamics) P1.1 program is a panel-method code which implements this integral equation for linearized subsonic flow in the complex-frequency domain, and is applicable to general shapes such as complete aircraft having arbitrary shapes, motions and deformations. Efficient computations are possible for multiple frequencies and multiple sets of vibration or deformation modes because the aerodynamic integrals are independent of both mode shapes and frequency and because the elements of the influence matrix depend on frequency in a very simple way. The P1.1 code employs the data base and data-handling utilities of the SPAR finite element structural analysis program (ref. 71). These were incorporated because SOUSSA P1.1 originally was intended for the calculation of steady-state structural loads and unsteady aerodynamics for flutter and gust-response calculation in multidisciplinary structural-optimization computations employing the SPAR structural analysis. The SPAR components, however, are unnecessary for stand-alone use. More efficient data handling methods for stand-alone operation are available.

Subsequent to the completion of the initial form of the SOUSSA P1.1 code several significant improvements have been incorporated and others are known to be needed (ref. 44). Among the latter are implementation of higher order panels (rather than assuming a constant velocity potential across the quadrilateral panels), elimination of the SPAR components, transposition and revision of the solution algorithm to substantially reduce input/output operations, and improved implementation of the trailing edge (Kutta) flow condition.

Some program improvements already incorporated in the SOUSSA code include the development of an "out-of-core" solver to permit the use of paneling schemes that lead to coefficient matrices too large to fit in the memory of modest-size computers; the replacement of the paneled wake by an analytical wake (reducing the cost of a typical run by about one-half) but retaining an option to use paneled wakes if needed (when there is another lifting surface in the wake); and replacing the rectangular panel integration by a Gaussian quadrature scheme to improve the accuracy of the calculated generalized aerodynamic forces. These improvements are incorporated in a replacement for the SOUSSA code (called UTSA) which is under development. In addition to the subsonic capability of the SOUSSA program, a supersonic proof-of-concept surface-panel code has been written to implement linear-theory algorithms developed under NASA grant (refs. 45 and 46). The code employs first-order panels and, like SOUSSA, is applicable to vehicles having arbitrary shapes, motions, and deformations. Validation and application of the code have begun.

For flow conditions that introduce nonlinearities in the equations (e.g. high subsonic and transonic flows) the integral-equation approach offers several advantages relative to finite-difference solution of the corresponding partial differential equation (PDE): (1) Evaluation of an integral is required rather than the numerical solution of a PDE, which is a more sensitive process. (2) The volume integral need be treated only in the relatively limited region of flow in which nonlinear terms are of significant magnitude rather than over the entire computational region as is common with finite-difference techniques. (3) Required accuracy can be attained with fewer computational grid points than would be required for finite-difference solution of the PDE. (4) Unlike current finite-difference solutions, the code is numerically stable even when moderate-to-large time steps are employed. (5) Correct far field boundary conditions are automatically imposed.

When nonlinearities are not negligible, the volume integral of the nonlinear terms is retained and integrated numerically, and the proof-of-concept transonic small perturbation computer program is called SUSAN (Steady and Unsteady Subsonic Aerodynamics - Nonlinear).

The SUSAN code has been limited to rectangular planforms and low frequencies. These limitations, however, were made for convenience and economy and are easily removed. Additional nonlinear terms could be readily included in the volume integral to solve the full potential equation. Initial applications have been to steady flow. Figure 12 shows chordwise pressure distribution near the root of a rectangular wing as calculated by the SUSAN code and by a transonic small perturbation finite difference code. The agreement is quite good even though only a few elements were used to evaluate the volume integral.

More recently the full potential equation has been solved in integral equation form for three-dimensional transonic flow at high angle of attack with vortex-type flow separation and shock waves (ref. 47). In this formulation, application of the Green's function method to the full-potential equation yields an expression for the velocity at any point in the flow as the sum of four parts, (1) freestream, (2) perturbation induced by the flow singularities representing the solid body (integral over the body surface), (3) perturbation induced by flow singularities representing the free vortex sheet representing the layered wakes from lifting-surface edges (integral over the vortex-sheet surfaces), and (4) the volume integral representing the compressibility (nonlinear) terms. The latter integrand decreases more rapidly than the square of the distance from the body or vortex surface, so the domain of integration can be relatively small. The sum of these four parts does not represent a superposition because the quantities in the integrands are not independent of each other. An iterative solution is employed to deform the free vortex sheets into a force-free shape.

Application of this method to the flow over an aspect-ratio-1.5 delta wing at freestream Mach number 0.7 is shown in figure 13 and compared with experiment, with a potential-flow calculation for incompressible flow, and with an Euler solution. The figure shows the shape of the free vortex sheet and Mach number contours as located at 0.8 of the root chord. In this exploratory calculation the vortex sheet was not allowed to roll up enough to exert its full inductive effect on the wing surface before the vorticity was transferred into the vortex core. If an additional quarter term of rollup were allowed, the pressure peak would be slightly higher and a little farther outboard, resulting in even better agreement with experiment. In contrast, the pressure peak from the Euler solution is considerably weaker and farther outboard than the experimental peak because of spatial and numerical diffusion in the Euler calculation.

To expedite viscous-flow computations, a scalar/vector-potential (SVP) formulation formally equivalent to Cauchy's equations of motion or the Navier-Stokes equations, has been developed for general three-dimensional, unsteady, compressible, viscous, heat-conducting flow (ref. 48). This formulation employs the classical Helmholtz decomposition of any vector field into the sum of an irrotational and a solenoidal field. The formulation is derived from fundamental principles of mechanics and thermodynamics. The governing equations for the scalar potential and vector potential have been obtained without restrictive assumptions on either the equation of state or the constitutive relations for the stress tensor and the heat flux vector. Since the outer region of the flow about an aircraft is essentially irrotational, an integral equation implementation is an especially attractive method of solution. A proof-of-concept code for two-dimensional flow is being developed and exercised.

Unsteady and Steady Pressure Measurements. - In parallel with unsteady aero code development/evaluation there has been a progressive unsteady pressure measurement program to provide an expanded data base for a variety of configurations, to permit code verification, and to help better understand flow field/body interaction phenomena. The experimental program was started several years ago and has progressed through several aerodynamic configurations. Four large wing models, shown in figure 14, have been tested in the LaRC Transonic Dynamics Tunnel (TDT) under the initial program (ref. 72).

The first model was a rigid highly swept, sharp leading edge, clipped delta planform configuration sidewall-mounted through a "splitter plate" and capable of being oscillated in pitch (ref. 73). Two control surfaces, one leading edge and one trailing edge, could be oscillated independently. Pertinent configuration parameters are given in the sketch in figure 15. An example of the static results is shown in figure 16. Calculated steady pressure from the Bailey-Ballhaus modified three-dimensional transonic small disturbance code (ref. 74) compared well with the static experimental data for wing angle of attack less than two degrees. At higher angles vortex flow from the sharp leading edge precludes satisfactory comparison since such flows are not accounted for in this code. Figure 17, which is an example of the oscillatory wing pitch results shows that the dominant features are the changes in the pressure magnitude induced by the motion of the mean shock, and the motion of the leading-edge vortex flow at four degrees angle of attack.

The second pressure model consisted of a half-body fuselage similar to that of a "wide body" transport and a rigid semispan supercritical wing representative of current energy efficient transport designs (fig. 14) (ref. 75). A sketch of the wing is shown in figure 18. The wing had 10 oscillating control surfaces. Only the inboard and outboard trailing edge surfaces indicated in figure 18 by the cross-hatched areas were used in the initial tests to generate unsteady airloads. Briefly, the measured results show that unsteady lifting pressures generated by oscillating control surfaces are substantial. In particular, the inboard oscillating control surface was shown to have a significant influence on the unsteady lifting pressure far outboard on the wing. Also, measured data were compared with calculated results obtained using a kernel-function subsonic lifting surface theory (ref. 76). An example of the results are presented in figure 19 which shows the chordwise distribution of lifting pressure caused by oscillating the inboard control surface at Mach 0.78. The measured pressures are not accurately predicted by the analyses over significant portions of the chord indicating a need for better prediction methods in the transonic speed range.

The planform and airfoil shape of the third rigid pressure model tested in the TDT is shown in figure 20. The unswept, rectangular planform, 12-percent thick wing was oscillated about the 0.46 chord. As shown in figure 21 steady and unsteady pressures were measured for a large number of test conditions (ref. 77). For the unsteady-data points (solid symbols) the wing oscillation frequencies were 5, 10, 15 and 20 hz. The Reynolds number based on chord length is 4 million for all the data.

The effect of the wing tip (that is, three-dimensional effects) on the pressure distributions was found to be large. Specifically, the shock location at the outboard sections were considerably further forward than for inboard sections. Mach number, as might be expected, also had a large effect on shock strength and location. Oscillation frequency had a significant effect on the magnitude and phase on the unsteady pressures.

Figure 22 is an example of comparisons of the measured data with calculated results from the XTRAN3S nonlinear transonic computer program and from the linear RHOIV kernel function program. In the region of the shock the calculations overestimated the leading edge pressures and underestimated them at the outboard station. The phase calculations are not in good agreement with measured values. The RHOIV calculations underestimate the unsteady pressure magnitude over the forward half of the wing and overestimate it over the aft portion. The shock, of course, is not predicted by the linear theory. The phase agreement is poor for both codes near the trailing edge.

The fourth wing tested was similar in planform to the second wing but differed significantly in that the second wing model was made as rigid as possible whereas the fourth wing was actually a very flexible wing off a drone vehicle used in the flight research program, Drones for Aerodynamic and Structural Testing (ref. 78). The wing planform and instrumentation are shown in figure 23 from reference 79. The wing was equipped with three trailing edge control surfaces but only the outboard surface shown in the figure was deflected statically and dynamically for the tunnel study. This wing also had a supercritical airfoil, varying from 15-percent thick at the root to 11-percent at the tip.

Steady and unsteady pressures and static and dynamic wing deflections were measured. Test conditions covered a wide range of Mach number from 0.60 to 0.90, dynamic pressures from less than 45 psf. to over 330 psf. Model parameter variations included wing angle of attack from -2 to 4 degrees, control surface mean deflection angle of -8 to 8 degrees and control surface oscillation amplitudes of 1, 2, and 3 degrees at frequencies of 5, 10, 15, and 20 Hz. Briefly, the steady pressure distributions show that span location, Mach number and angle of attack have a large effect on the mean shock strength and chordwise location. The unsteady pressure distributions show that large peaks in the pressure magnitude occur due to both the oscillatory control surface and to the motion of the mean shock location. Frequency effects were shown to be non-linear and exceedingly large if the oscillatory frequency occurs near a natural mode of the wing structure. Static tip deflections were large (4 in. of vertical deflection and 3 degrees of twist) and nonlinear with increasing dynamic pressure.

The results of these four pressure model studies are summarized in more detail in ref. 72.

The initial unsteady pressure measurement program has been carried a step further with tests of a two-dimensional rigid supercritical airfoil model at Reynolds numbers up to 35 million in the Langley Research Center 0.3m Transonic Cryogenic Tunnel (TCT). The model and some preliminary results are shown in figure 24. The objectives of the tests were to determine Reynolds number effects on unsteady pressures, increase the data base for developing computer codes at operational Reynolds numbers, and to develop techniques for cryogenic pressure measurements. The flow test conditions are shown in figure 22. The model was oscillated in pitch at frequencies from 5 to 60 Hz with oscillation amplitudes from 1/4 to 1.0 degree at mean angles of attack from -2.5 to 3.0 degrees. Both wing surface and wake pressure measurements were made. The data currently is being analyzed. In addition to providing new data the tests provided an opportunity to address some of the known and unknown problems associated with dynamic testing at cryogenic temperatures. This study is considered a precurser to future desirable tests at high Reynolds numbers including an oscillating control surface test in the 0.3m TCT and a three-dimensional wing oscillating in pitch in the LaRC National Transonic Facility.

RECENT ACCOMPLISHMENTS IN THE EXPERIMENTAL PREDICTION OF AEROELASTIC CHARACTERISTICS AT THE NASA LANGLEY RESEARCH CENTER

Tests several years ago in the TDT of two structurally identical transport type wings but with different airfoils ("supercritical" and "conventional") showed the supercritical wing to have a lower transonic flutter boundary than the conventional wing (ref. 80). In addition, the minimum flutter speed was ill-defined and apparently a function of angle of attack. A later study investigated the effect of root angle of attack, α_0 on flutter (ref. 81). Some of the results of this study are shown in figure 25. Near the design Mach number of 0.98, the flutter boundary, shown on the left of the figure as flutter speed index versus Mach number, was found to curl backwards for angles-of-attack above zero. This curlback occurs for a range of Mach numbers of 0.95 to 1.0. The bottom of the transonic dip could not be determined for $\alpha_0 = 2°$ because of the difficulty in reducing tunnel pressures to lower values.

Since classical unsteady aerodynamic flutter solutions do not predict this novel behavior, an analytical study was conducted to see if a recently developed transonic finite-difference computer code would exhibit the phenomenon. Flutter boundaries of two-dimensional airfoils were calculated using the small perturbation theory HYTRAN2 code for a range of Mach numbers and angles of attack, α. The airfoil studied was the supercritical MBB A-3. The sketch on the right side of figure 25 shows a two-dimensional airfoil section mounted on a pitch spring. This simple model is analogous to the effect of washout of angle of attack at the tip of a loaded wing. The static nosedown pitching moment, c_m, twists the section from its "root" angle-of-attack, α_0, to a smaller angle, α. The amount of twist is proportional to the total moment and thus to dynamic pressure. When the flutter speed index is plotted versus Mach number for $\alpha = 0$ and 1 deg., a significant transonic dip is seen but there is no evidence of the curl back seen in the TF-8A results. When the effect of static twisting is included and the boundary plotted for "root" angles, α_0, a curl back develops between 2 and 4 degrees. The curl back is due to the static twisting of the airfoil under the combined influences of Mach number and dynamic pressure. Since the flutter speed index is proportional to the square root of dynamic pressure, the nosedown twist angle, $\alpha-\alpha_0$, decreases as the flutter speed index decreases. The resulting higher angles, α, near the bottom of the transonic dip induced transonic effects which produce the curl back of the flutter boundary.

Since adequate theories for 3D unsteady transonic flow are just now being developed and verified, flutter calculations corresponding to the test conditions and have been made by a modified strip analysis which requires as input spanwise distributions of section lift-curve slope and aerodynamic center. These aerodynamic parameters were obtained from pressure distributions measured in the 8-Foot Transonic Pressure Tunnel with a model that was geometrically similar to but much stiffer than the flutter model.

The results are shown in figure 26 (ref. 81) where the experimental data of figure 25 also is repeated. Flutter calculations for values of the mass ratio μ which bracket the experimental values are slightly conservative but in good agreement with the experimental boundary at α = 0°. At nonzero angle of attack, the calculations do not show the backward turn but indicate instead a double dip similar to that observed in some earlier tests in TDT and in supercritical wing flutter tests recently completed at the Netherlands NLR. It is believed that the backward turn was not calculated primarily because the aerodynamic model was two orders of magnitude stiffer than the flutter model and hence did not deform nearly as much as the flutter model. Thus, the disagreement between the calculated and experimental results at α = 2° is attributed at least in part to differences in mean deformed shape between aerodynamic and flutter models. Earlier calculations for α, χ, o, for which static aeroelastic deformation was minor, has shown excellent agreement with the experimental flutter boundary (ref. 33).

Angle of attack effects also figured prominently in unusual aeroelastic characteristics of a forward swept wing configuration tested in the TDT (ref. 82). There has been increasing evidence that angle of attack can be a significant parameter relative to aeroelastic stability of aerodynamically efficient configurations at high subsonic/transonic speeds. In addition to the studies previously mentioned other examples have been encountered. Unexpected oscillations of the canard lifting surface of a statically-scaled aeroelastic model of the HIMAT (Highly Maneuverable Aircraft Technology) demonstrator drone aircraft (ref. 83) occurred over a limited angle of attack and Mach number range during static loads testing in the wind tunnel, and limit cycle oscillations of the B-1 wing occurred during flight test maneuvers (ref. 84). The latter phenomenon has been studied extensively both analytically and in wind tunnel studies in the Langley TDT and at the Ames Research Center (refs. 85, 86). It has been hypothesized that the oscillations result from unsteady shock/boundary layer interaction or from unsteady leading edge vortex flow at critical angles of attack but further studies apparently are in order to better define the phenomena.

Another unusual flutter boundary recently was encountered in the unsteady pressure tests of the high aspect ratio DAST ARW-2 wing mentioned previously. The results of the tests are shown in figure 27 which, in addition to the pressure measurement tunnel conditions, shows an unexpected flutter instability in the transonic region that was encountered at nearly constant Mach number (in both air and freon test media) extending from very low dynamic pressure to near the dynamic pressure limit of the tunnel. The wing motion was predominantly wing bending. The frequency of this mode varied from 8.3 Hz wind-off to 13 Hz at the highest dynamic pressure. The predicted flutter frequency using linear theory was 24 Hz (at 0.8 Mach number). As shown in figure 27 these linear calculations predict flutter at considerably higher dynamic pressure than was encountered during the test. Additional testing of the wing is planned to further define the mechanism of the instability, to determine the effectiveness of the active control system in controlling the instability and to extend the measured instability boundary to higher dynamic pressure making use of the increased operational capability recently provided to the TDT.

In another study in the TDT an unusual "body-freedom flutter" (BFF) was demonstrated on a model of a forward swept wing (FSW). This phenomenon is caused by adverse coupling of rigid-body-pitching and wing-bending motions. Although rare on aft-swept-wing aircraft, this mechanism is generic to FSW configurations due to the tendency of the wing effectively to "destiffen" with increasing dynamic pressure (leading eventually to wing failure by divergence if flutter is not encountered first). Because this flutter mode was of concern for the X-29A aircraft a half-scale semi-span model of an early FSW aircraft design was tested on a cable and rod sidewall mount system that provided the necessary pitch and plunge degrees of freedom (ref. 87). In addition, the model was equipped with a stability augmentation system (SAS) that allowed the comparison of BFF speeds of statically stable and unstable configurations. A hydraulically actuated canard was used to provide the SAS forces. Some results of these tests are presented in figure 28 which shows calculated and experimentally predicted BFF boundaries. The aeroservoelastic analysis which incorporated doublet-lattice aerodynamics modified by static aeroelastic test data adequately predicts the BFF boundaries at the lower Mach numbers, but is unconservative at the higher Mach numbers.

An example of the need to be sensitive to the effects on aeroelastic stability and response of innovative aerodynamic configurations that bring increased aerodynamic efficiency is illustrated by a series of studies in the TDT of the effects of "winglets" on wing flutter. Sidewall-mounted semi-span models of three different transport-type swept-tapered wing configurations were tested: a clean-wing of an executive transport jet aircraft (ref. 88); a large twin-engine (wing-mounted) transport wing (ref. 89); and an advanced large four engine transport wing (ref. 90). Each wing was tested with a normal (clean) wing tip, a tip-mounted winglet, and a tip ballasted to simulate the mass and inertia properties of the winglets. In all cases it was shown that the flutter speed of the basic wing was reduced by the aerodynamic effects of added winglets - but that the amount of reduction was very configuration dependent. The bare wing experienced a slight (2 percent) reduction in flutter speed due aerodynamic effects; the single-engined wing a 20-percent decrease (see figure 29 for example); and the twin-engined wing a 15-percent decrease.

The problem of predicting the flutter characteristics of aircraft carrying external stores (fuel tanks, bombs, rockets, ECM pods, etc.) has been highlighted in recent years by the number of flutter studies in the TDT that has been required for the F-16 fighter. In addition to the many stores flutter clearance tests for the basic F-16 fighter (see figure 30) and more recent tests of the advanced F-16E configuration there have been numerous tests of a variety of new external stores for the F-16. These included tests of the effects of new 500 gal fuel tanks and non-jettisonable multipurpose pylons (some sample stores are shown in figure 28), and new air-to-air missile configurations. The analytical predictions did not always agree with the experimental results. These studies are discussed more fully in references 81 and 91.

RECENT ACCOMPLISHMENTS IN ACTIVE CONTROL OF AEROELASTIC STABILITY AND RESPONSE AT THE NASA LANGLEY RESEARCH CENTER

One of the more recent active control studies in the TDT utilized a modified version of the General Dynamics built 1/4-scale F-16 flutter model (fig. 30). A duplicate set of wings which permitted the use of the "flaperon" as an active control surface, and an on-board miniature hydraulic control system were required for the model (ref. 92). These tests successfully demonstrated that active controls could be used to negate the decrease in both the symmetric and antisymmetric flutter speeds often brought about by the carriage of external stores.

The unique model shown in figure 31 has been used in several studies in the TDT to generate much useful information on suppression of wing/store flutter with active controls. The Northrop built semi-span 30-percent scale flutter model, simulating the YF-17 airplane, is sidewall-mounted on a system of bars and cables that allows the flexible half-fuselage to pitch and plunge (ref. 93). The most recent of a series of tests progressing from analog computer, non-adaptive active control to digital computer, adaptive control demonstrated the feasibility of quick adaptive capability by ejecting a store the absence of which caused the wing to go unstable. The active control system quickly adapted to the new configuration and damped out the instability.

In addition to the wind tunnel and full-scale aircraft studies, a unique flight research program called DAST (Drones for Aerodynamic and Structural Testing) (ref. 94) provided a focus for evaluation and improvement of synthesis and analysis procedures for design of active control systems on wings with significant aeroelastic effects. Two different research wings were designed to replace the standard wing of an Air Force version of the Firebee II target drone which was used as the remotely piloted test bed. The first wing, ARW-1, was designed to validate a flutter suppression system and aeroelastic effects on aerodynamic loads at transonic speeds. The primary objective of the second wing, ARW-2 was to evaluate multiple active controls systems operating simultaneously, the proper functioning of which are necessary to preserve structural integrity for various flight conditions. The design of ARW-2 involved what is believed to be the first exercise of an iterative procedure integrating aerodynamics, structure, and controls technologies in a design loop resulting in flight hardware. Much was learned of the practical design and synthesis of flutter suppression systems in the design and flight testing of ARW-1 and the design and fabrication of ARW-2. The program was terminated due to severe funding restrictions before ARW-2 could be flight tested. Consequently, the wing will be used for further wind tunnel studies.

Although these series of tests demonstrated the effectiveness of the active controls concept for suppressing wing/store flutter, the gain and phase settings determined from analysis before the tests in many instances had to be changed during the tests to obtain near optimum performance, and the need for better analytical methods was indicated. The analytical methods, both analysis and design, have been evolving for some time. In design, the problem is the synthesis of a control law that regulates the dynamics of the system such that measures of performance (e.g. stability, loads, transient response, etc.) are acceptable. In analysis, the control law already exists and analytical methods are employed to assess the performance of the overall controlled system. In reference 95, which reviews analytical methods and associated tools for active controls analysis and design problems, it is emphasized that to take full advantage of active control technology control law synthesis and analysis must be an integral part of the aircraft design process. This requires efficient methods and accompanying tools that will enable the active controls designer routinely to synthesize and analyze complex control systems.A significant amount of work has been performed to develop analysis and synthesis methods for this task. Much of the NASA active controls research involvement has been at the forefront of these developments. Some of this work is summarized in reference 96, excerpts of which will be repeated here to give a flavor of the activity.

The analysis of an actively controlled flexible aircraft requires that the interfaces among unsteady aerodynamics, structures, and control theory be properly considered. Because of the multidisciplinary nature of the problem, the format of the equations of motion and the analytical methods used to solve them are many times inconsistent. To properly handle these problems is a complex task that requires the use of efficient multidiscipline computer programs. The need for these types of computer programs was evident in the early 70's and as a result several computer programs for analyzing actively controlled flexible aircraft were developed. One of the first of these programs was FCAP (Flight Controls Analysis Program) developed under contract by Aerospace Systems, Inc. (ref. 97). The ISAC (Interaction of Structures, Aerodynamics, and Controls) program was developed at Langley and is used regularly in NASA-related research (ref. 98). DYLOFLEX is an integrated system of stand-alone computer programs which performs dynamic loads analyses of flexible airplanes with active controls (ref. 99). It was developed under contract by the Boeing Company and is available from COSMIC (Computer Software Management and Information Center).

All of these programs incorporate unsteady aerodynamics. From a stability calculation point of view, the incorporation of unsteady aerodynamics presents a problem that each program handles differently. Figure 32 illustrates a solution to this problem that has received considerable attention. Unsteady aerodynamics are computed for simple harmonic motion at specified values of frequency. Futhermore, the real and imaginary parts of the aerodynamic forces are available only in tabular form ($Q(i\omega)$). The approach taken here and implemented in the ISAC program is to allow the variation of the aerodynamic forces with frequency to be approximated by a rational function of $i\omega$ ($f(i\omega)$) (ref. 100). There are several techniques available for obtaining a rational function. The most widely known are the so-called least squares method and the Pade method. With either of these methods, the Laplace variable s is substituted for $i\omega$ and time derivatives are then associated with the powers of s. This results in a set of constant coefficient differential equations that can be used in an eigenvalue analysis to determine stability. In addition, the equations of motion are in a form that can be used for control law design which also has been an integral part of LaRC research in the control of aeroelastic stability and response. Since control law design generally is more difficult than analysis, it has lagged behind from the viewpoint of production computer programs. The primary emphasis of work at the LaRC Loads and Aeroelasticity Division has been the development of design methodology.

Three methods for designing active control systems have been applied to the flutter suppression problem (refs. 101 and 102) namely, the "classical," "aerodynamic theory," and "optimal control theory" methods. These basic methods are now being applied to other active control functions (ref. 103). Optimal control theory seems to be the best suited for the task of designing multifunction active control systems. Therefore, considerable attention is being given to design methods that employ this approach. Optimal control theory is based on the design of a controller which minimizes a performance function (such as control deflection, bending moment, acceleration, etc). The usual method for designing a low-order control law from optimal control theory is to approximate the full-order control law through order reduction techniques such as truncation, residualization, and transfer function matching. These techniques all result in low-order control laws that are not optimal.

A new approach has been developed (ref. 104) that begins with a full-order controller. Using engineering judgment, a few key states and their associated design variables and initial values are selected from the full-order solution. A nonlinear programing algorithm is then used to search for the values of the control law variables which minimize the performance function. The resulting low-order control law is optimal for the states selected. The method is direct and results in a control law that is much easier to implement in a flight computer.

References

1. Goland, Martin: An Appraisal of Aeroelasticity in Design with Special Reference to Dynamic Aeroelastic Stability. Pres. At Sixth Anglo-American Aeronautical Conference, London, England, September 1-15, 1957.

2. Schier, J. F. and Juergens, R. J.: Design Impact of Composites in Fighter Aircraft - Part 1: They Force a Fresh Look at the Design Process. Astronautics and Aeronautics, Sept. 1983.

3. Demario, William F.: New World for Aerospace Composites. Aerospace America, October 1985.

4. Sliwa, Stephen M. and Arbuckle, P. Douglas: Implications of Control Technology on Aircraft Design. In NASA Aircraft Controls Research, NASA CP 2296, October, 1983.

5. Shomber, H.: Application of Integrated Active Controls in Future Transports. AIAA Paper 79-1654, Aug. 1979.

6. Edwards, J. W.; Bland, S. R.; and Seidel, D. A.: Experience with Transonic Unsteady Aerodynamic Calculations. NASA TM 86278, Aug. 1984.

7. Weisshaar, Terrence A.; and Foist, Brian L.: Aeroelastic Tailoring of Aircraft Subject to Body Freedom Flutter. AFWAL-TR-83-3123, Nov. 1983.

8. Many authors: Recent Experiences in Multidisciplinary Analysis and Optimization. NASA CP 2327, Proceedings of a Symposium held at Hampton, VA April 24-26, 1984.

9. Anon: Aeronautics Technology Possibilities for 2000 - Report of a Workshop by Aeronautics and Space Engineering Board, Commission on Engineering and Technical Systems, National Research Council. National Academy Press, Wash. D.C., 1984.

10. Bendixen, G. E.; O'Connell, R. F.; and Siegert, C. D.: Digital Active Control System for Load Alleviation for the Lockheed L-1011. Aeronautical Journal, Vol. 85, Nov. 1981, pp. 430-436.

11. Sensburg, O.; Becker, J.; Lusebrink, H.; and Weiss, F.: Gust Load Alleviation on the Airbus A300. ICAS Paper 82-211, Seattle, Wash., Aug. 1982.

12. DeMeis, Richard: Joined Wing - Child of the Computer. Aerospace America, April, 1985.

13. Piccirillo, Albert C.: The Advanced Tactical Fighter - Design Goals and Technical Challenges. Aerospace America, Nov. 1984.

14. Gregory, Tom: Oblique Wing Ready for Research Aircraft. Aerospace America, June 1985.

15. Holt, Daniel: X-29 Testing Progresses. Aerospace Engineering, March 1985.

16. Garrick, I. E. and Reed, W. H. III: Historical Development of Flutter. Journal of Aircraft, Vol. 18, No. 11, Nov. 1981.

17. Collar, A. R.: The First Fifty Years of Aeroelasticity. Aerospace, Vol. 5, (Paper No. 545), Feb. 1978, pp. 12-20.

18. Reed, Wilmer H. III: Aeroelasticity Matters: Some Reflections on Two Decades of Testing in the NASA Langley Transonic Dynamics Tunnel. NASA TM 83210, Sept. 1981.

19. Prandtl, L.: Wing Theory in a Compressible Medium. Luftfahrtforschung, Vol. 13, Oct. 1936, pp. 313-319.

20. Kussner, H. G.: General Airfoil Theory. Luftfahrtforschung. Vol. 17, 1940, pp. 470-478. Also NACA TM-979, Dec. 1941.

21. Watkins, C. E.; Runyan, H. L.; and Woolston, D. S.: On the Kernel Function of the Integral Equation Relating the Lift and Downwash Distributions of Oscillating Finite Wings in Subsonic Flow. NACA TN-3131, 1954. Also NACA TR 1234, 1955.

22. Yates, E. Carson, Jr.: A Kernel-Function Formulation for Nonplanar Lifting Surfaces Oscillating in Subsonic Flow. AIAA Journal, Vol. 14, No. 8, Aug. 1966.

23. Watkins, Charles E.; Woolston, Donald S.; and Cunningham, Herbert J.: A Systematic Kernel Function Procedure for Determining Aerodynamic Forces on Oscillating or Steady Finite Wings at Subsonic Speeds. NASA TR R-48, 1959.

24. Albano, Edwards and Rodden, William P.: A Doublet Lattice Method for Calculating Lift Distributions on Oscillating Surfaces in Subsonic Flows. AIAA Journ. Vol. 7, No. 2, Feb. 1969, pp 279-285; Errata, Vol. 7. No. 11, Nov. 1969, p. 2192.

25. Jones, W. P.: Aerodynamic Forces on Wings in Non-Uniform Motion. R and M 2117, 1945.

26. Morino, L.: A General Theory of Unsteady Compressible Potential Aerodynamics. NASA CR-2464, Dec. 1974.

27. Suciu, E. O. and Morino, L.: Nonlinear Finite Element Analysis of Lifting Surfaces in Incompressible Flow with Wake Roll-up. AIAA Paper 76-64. Pres. at AIAA 14th Aerospace Sciences Meeting, Washington D.C., Jan. 26-28, 1976.

28. Morino, L. and Tseng, Kadin: Time-Domain Green's Function Method for Three-Dimensional Nonlinear Subsonic Flows. AIAA Paper 78-1204. Pres. at AIAA 11th Fluid and Plasmadynamics Conference, Seattle, Washington, July 10-12, 1978.

29. Pines, Samuel; Dugundji, John; and Neuringer, Joseph: Aerodynamic Flutter Derivatives for a Flexible Wing with Supersonic and Subsonic Edges. J. Aeronautical Sci., vol. 22, no. 10, Oct. 1955, pp 693-700.

30. Cunningham, Herbert J.: Application of a supersonic Kernel-function Procedure to Flutter Analysis of Thin Lifting Surfaces. NASA TN D-6012, 1970.

31. Yates, E. Carson, Jr.: Calculation of Flutter Characteristics for Finite-Span Swept or Unswept Wings at Subsonic and Supersonic Speeds by a Modified Strip Analysis. NACA RM L57L10, 1958.

32. Yates, E. Carson, Jr.: Modified-Strip-Analysis Method for Predicting Wing Flutter at Subsonic to Hypersonic Speeds. Journal of Aircraft, Jan.-Feb., 1966.

33. Yates, E. C., Jr; Wynne, E. C.; Farmer, M. G; and Desmarais, R. N.: Prediction of Transonic Flutter for a Supercritical Wing by Modified Strip Analysis. Journal of Aircraft, Nov. 1982.

34. Yates, E. C., Jr.; Wynne, E. C.; and Farmer, M. G.: Effects of Angle of Attack on Transonic Flutter of a Supercritical Wing. Journal of Aircraft, Oct. 1983.

35. Bland, Samuel R.: Recent Advances and Concepts in Unsteady Aerodynamic Theory. In Aerodynamic Analyses Requiring Advanced Computers - Part 2, NASA SP-347, March 1975.

36. Ehlers, F. Edward: A Finite Difference Method for the Solution of the Transonic Flow Around Harmonically Oscillating Wings. NASA CR-2257, Jan. 1974.

37. Ballhaus, W. F. and Goorjian, P. M.: Implicit Finite-Difference Computations of Unsteady Transonic Flow About Airfoils. AIAA Journal, Vol. 15, no. 12, Dec. 1977, pp. 1728-1735.

38. Staff of the Aeroelasticity Branch, LaRC: The Langley Transonic Dynamics Tunnel. LWP-799, Sept. 1969.

39. Magnus, Alfred E. and Epton, Michael E.: PANAIR - A Computer Program for Predicting Subsonic or Supersonic Linear Potential Flows about Arbitrary Configurations Using a Higher Order Panel Method. Volume I - Theory Document. NASA CR 3251, 1980.

40. Morino, Luigi: Steady, Oscillatory, and Unsteady Subsonic and Supersonic Aerodynamics - Production version (SOUSSA P1.1) - Vol. I - Theoretical Manual. NASA CR 159130, 1980.

41. Tseng, K.: Nonlinear Green's Function Method for Transonic Potential Flow. Ph.D. Dissertation, Boston University, 1983.

42. Whitlow, W. Jr.: XTRAN2L: A Program for Solving the General Frequency Unsteady Transonic Small Disturbance Equation. NASA TM 85723, Nov. 1983.

43. Borland, C. J. and Rizzetta, D. P.: Nonlinear Transonic Flutter Analysis. AIAA Journal, Vol. 20, No. 11, Nov. 1982, pp. 1606-1615.

44. Yates, E. Carson, Jr.; Cunningham, Herbert J.; Desmarais, Robert N.; Silva, Walter A.; and Drobenko, Bohdan: Subsonic Aerodynamic and Flutter Characteristics of Several Wings Calculated by the SOUSSA P1.1 Panel Method. NASA TM 84485, 1982, Also AIAA 82-0727.

45. Freedman, Marvin I.; Sipcic, Slobodan; and Tseng, Kadin: A First-Order Green's Function Approach to Supersonic Oscillatory Flow - A Mixed Analytical and Numerical Treatment. NASA CR 172207, Feb. 1985.

46. Freedman, Marvin I. and Tseng, Kadin: A First-Order Time-Domain Green's Function Approach to Supersonic Unsteady Flow. NASA CR 172208, April 1985.

47. Kandil, Osama A. and Yates, E. Carson, Jr.: Computation of Transonic Vortex Flows Past Delta Wings - Integral Equation Approach. AIAA Paper No. 85-1582. Presented at AIAA 18th Fluid Dynamics and Plasmadynamics and Lasers Conference, Cincinnatti, Ohio, July 16-18, 1985.

48. Morino, Luigi: Scalar/Vector Potential Formulation for Compressible Viscous Unsteady Flows. NASA CR 3921, Aug. 1985.

49. Ricketts, Rodney H.: Selected Topics in Experimental Aeroelasticity at the NASA Langley Research Center. NASA TM 86436, April 1985.

50. Hanson, Perry W.: An Aeroelasticians' Perspective of Wind Tunnel and Flight Experiences with Active Control of Structural Response and Stability. NASA TM 85761, April 1984.

51. Sandford, Maynard C.; Abel, Irving; and Gray, David L.: Development and Demonstration of a Flutter Suppression System Using Active Controls. NASA TR R-450, Dec. 1975.

52. Redd, L. T.; Gilman, J., Jr.; Cooley, D. E.; and Severt, F. D.: Wind Tunnel Investigation of a B-52 Model Flutter Suppression System. Journal of Aircraft, Vol. II, No. 11, Nov. 1974.

53. The Boeing Company: B-52 CCV Program Summary. AFFDL-TR-74-92, March 1975.

54. Doggett, Robert V., Jr.; Abel, Irving; and Ruhlin, Charles L.: Some Experiences Using Wind Tunnel Models in Active Control Studies. NASA TM-X 3409, Aug. 1976.

55. Disney, T. E.: C-5A Load Alleviation. Active Controls in Aircraft Design. AGARD CP-234, Nov. 1978.

56. Abel, I. and Newsom, J. R.: Wind-Tunnel Evaluation of NASA-Developed Control Laws for Flutter Suppression on a DC-10 Derivative Wing. AIAA Paper No. 81-0639, April 1981.

57. Kandil, Osama A.; Chu, Li-Chuan; and Yates, E. Carson, Jr.: Hybrid Vortex Method for Lifting Surface With Free-Vortex Flow. AIAA Paper 80-0070, 1980.

58. Many Authors: Unsteady Airloads in Separated and Transonic Flow. AGARD SMP CP-374. 59th Meeting of AGARD Structures and Materials Panel, Toulouse, France, Sept. 3-7, 1984.

59. Hanson, Perry W.: The Prediction of Buffet Flow: A State-of-the-Art Review. NASA TM X-72627, Dec. 1974.

60. Butler, G. F. and Jones, J. G.: The Prediction of Buffeting Response in Flight from Wind-Tunnel Measurements on Models of Conventional Construction. Aero. Journal of the Royal Aeronautical Society, Aug/Sept. 1984.

61. Cunningham, A. M., Jr. and Benepe, D. B., Sr.: Prediction of Transonic Aircraft Buffet Response. AGARD CP-226, July 1977.

62. Hanson, P. W.: Evaluation of an Aeroelastic Technique for Predicting Airplane Buffet Loads. NASA TN D-7066, 1973.

63. Triplett, William E.: Pressure Measurements on Twin Vertical Tails in Buffeting Flow. Vol. 1 - General Description. AFWAL-TR-82-3015, April 1982.

64. Seidel, D. A.; Bennett, R. M.; and Whitlow, W., Jr.: An Exploratory Study of Finite Difference Grids for Transonic Unsteady Aerodynamics. NASA TM 84593, Dec. 1982.

65. Edwards, J. W.; Bland, S. R.; and Seidel, D. A.: Experience with Transonic Unsteady Aerodynamic Calculations. NASA TM 86278, August 1984.

66. Bennett, Robert M.; Wynne, Eleanor C.; and Mabey, Dennis G.: Calculation of Transonic Steady and Oscillatory Pressure on a Low Aspect Ratio Model and Comparison with Experiment. NASA TM 86449, June 1985.

67. Fuglsang, D. F. and Williams, M. H.: Non-Isentropic Unsteady Transonic Small Disturbance Theory. AIAA Paper 85-0600. Pres. AIAA/ASME/ASCE/AHS 26th Structures, Structural Dynamics and Materials Conf., Orlando, Florida, April 15-17, 1985.

68. Howlett, J. T.: Effective Self-Consistent Viscous-Inviscid Solutions for Unsteady Transonic Flow. NASA TM 86335, Jan. 1985.

69. Batina, J. T.: Unsteady Transonic Flow Calculations for Two-Dimensional Canard-Wing Configurations with Aeroelastic Applications. NASA TM 86375, Feb. 1985.

70. Batina, John T.: Unsteady Transonic Flow Calculations for Interfering Lifting Surface Configurations. NASA TM 86432, July 1985.

71. Giles, G. L. and Haftka, R. T.: SPAR Data Handling Utilities. NASA TM-78701, 1978.

72. Sandford, M. C.; Ricketts, R. H.; and Hess, R. W.: Recent Transonic Unsteady Pressure Measurements at the NASA Langley Research Center. NASA TM 86408, April 1985.

73. Hess, R. W.; Wynne, E. C.; and Cazier, F. W., Jr.: Static and Unsteady Pressure Measurements on a 50-Degree Clipped Delta Wing at M = 0.9. NASA TM 83297, April 1982.

74. Mason, W. H.; Ballhaus, W. F.; Mackenzie, C.; Frick, J.; and Stern, M.: An Automated Procedure for Computing the Three-Dimensional Transonic Flow Over Wing-body Combinations, Including Viscous Effects. AFFDL-TR-77-122.

75. Sandford, Maynard C.; and Ricketts, Rodney H.: Steady- and Unsteady-Pressure Measurements on a Supercritical Wing Model with Oscillating Control Surfaces at Subsonic and Transonic Speeds. NASA TM-84543, 1983.

76. Rowe, W. S.; Sebastian, J. D.; and Petrarca, J. R.: Reduction of Computer Useage Costs in Predicting Unsteady Aerodynamics Loadings Caused by Control Surface Motions - Analysis and Results. NASA CR 3009, March 1979.

77. Ricketts, R. H.; Sandford, M. C.; Watson, J. J.; and Seidel, D. A.: Subsonic and Transonic Unsteady- and Steady-Pressure Measurements on a Rectangular Supercritical Wing Oscillated in Pitch. NASA TM-85765, July 1984.

78. Murrow, H. N. and Eckstrom, C. V.: Drones for Aerodynamic and Structural Testing (DAST) - A Status Report. Journ. of Aircraft, Vol. 16, No. 8, Aug. 1979, pp. 521-526.

79. Seidel, David A.; Sandford, Maynard M.; and Eckstrom, Clinton, V.: Measured Unsteady Transonic Aerodynamic Characteristics of an Elastic Supercritical Wing with an Oscillating Control Surface. NASA TM , April 1985.

80. Farmer, Moses G. and Hanson, Perry W.: Comparison of Supercritical and Conventional Wing Flutter Characteristics. NASA TM X-72837, May 1976.

81. Gardner, James E.: Loads and Aeroelasticity Division Research and Technology Accomplishments for FY 1982 and Plans for FY83. NASA TM 84594, Jan. 1983.

82. Doggett, Robert V., Jr.; and Ricketts, Rodney H.: Dynamic Response of a Forward-Swept-Wing at Angles of Attack up to 15° at a Mach Number of 0.8, NASA TM 81863, Nov. 1980.

83. Lockenour, Jerry L. and Layton, Garrison P.: RPRV Research Focus on HIMAT. Astronautics and Aeronautics, April 1976, pp. 36-41.

84. Stevenson, J. R.: Shock-Induced Self-Excited Airfoil Bending Oscillations. Paper presented to Aerospace Flutter and Dynamics Council, Los Vegas, Nevada, Oct. 19-20, 1978.

85. Stevenson, J. R.: Shock-Induced Wing Oscillation Wind Tunnel Test Results. Paper presented to Aerospace Flutter and Dynamics Council, Williamsburg, Va, April 4-6, 1984.

86. Dobbs, S. K.; Miller, G. D.; and Stevensen, J. R.: Self Induced Oscillation Wind Tunnel Test of a Variable Sweep Wing. AIAA/ASME/ASCE/AHS 26th Structures, Structural Dynamics and Materials Conf., Orlando, Florida, April 15-17, 1985. AIAA Paper No. 850739-CP, 1985.

87. Chipman, Richard; Rauch, Frank; Rimer, Melvyn; Muniz, Benigno; and Ricketts, Rodney H.: Transonic Tests of a Forward-Swept-Wing Configuration Exhibiting Body-Freedom Flutter. Proceedings of the AIAA/ASME/ASE/AHS 26th Structures, Structural Dynamics, and Materials Conf. Orlando, Fla., April 15-17, 1985. AIAA Paper No. 85-0689-CP, 1985.

88. Ruhlin, C. L.; Rauch, F. J.; and Waters, C.: Transonic Flutter Model Study of a Supercritical Wing and Winglet. Journal of Aircraft, Vol. 20, No. 8, August, 1983.

89. Bhatia, Kumar G.; Nagaraja, K. S.; and Ruhlin, Charles E.: Winglet Effects on the Flutter of Twin-Engine-Transport-Type Wing. Proc. of AIAA/ASME/ASCE/AHS 25th Structures, Structural Dynamics, and Materials Conference, Palm Springs, CA, May 14-16, 1984. AIAA Paper No. 84-0905-CP, May 1984.

90. Pearson, R. M.; Geising, J. P.; Nomura, J. K.; and Ruhlin, C. L.: Transonic Flutter Model Study of a Multi-Jet Aircraft Wing with Winglets. Proceedings of AIAA/ASME/ASCE/AHS 26th Structures, Structural Dynamics, and Materials Conference, Orlando, Fla, April 15-17, 1985. AIAA Paper No. 85-0738-CP, 1985.

91. Gardner, James E. and Dixon, S. C.: Loads and Aeroelasticity Division Research and Technology Accomplishments for FY 1984 and Plans for FY 1985. NASA TM 86356, Jan., 1985.

92. Peloubet, R. P., Jr.; Haller, R. L.; and Bolding, R. M.: Recent Developments in the F-16 Flutter Suppression with Active Controls Program. Pres. at AIAA/ASME/ASCE/AHS Structures, Structural Dynamics, and Materials Conf, May 2-4, 1983, Lake Tahoe, Nev., AIAA Paper No. 83-995.

93. Johnson, E. H.; Hwang, C.; Pi, W. S.; Kesler, D. F.; Joshi, D. S.; and Harvey, C. A.: Test Demonstration of Digital Control of Wing/Store Flutter. AIAA CP 82-0645, May 82.

94. Murrow, H. N. and Eckstrom, C. V.: Drones for Aerodynamic and Structural Testing (DAST) - A Status Report. Journal of Aircraft, Vol. 16, No. 8, August 1979, pp. 521-526.

95. Newsom, J. R.; Adams, W. M., Jr.; Mukhopadhay, V.; Tiffany, S. H.; and Abel, I.: Active Controls - A Look at Analytical Methods and Associated Tools. Pres. at 14th Congress of the International Council of the Aeronautical Sciences (ICAS), Toulouse, France, Sept. 10-14, 1984.

96. Newsom, Jerry R. and Abel, Irving: Some Experiences with Active Control of Aeroelastic Response. NASA TM 83179, July 1981.

97. Noll, R. B. and Morino, L.: FCAP - A New Tool for Evaluation of Active Control Technology. AIAA Paper No. 75-1059, Aug. 1975.

98. Peele, Elwood L. and Adams, William M., Jr.: A Digital Program for Calculating the Interaction Between Flexible Structures, Unsteady Aerodynamics, and Active Controls. NASA TM 80040, 1979.

99. Perry, B. III; Kroll, R. I.; Miller, R. D.; and Goetz, R. C.: DYLOFLEX: A Computer Program for Flexible Aircraft Dynamic Loads Analyses with Active Controls. J. Aircraft, Vol. 17, No. 4, April 1980.

100. Abel, I.: An Analytical Technique for Predicting the Characteristics of a Flexible Wing Equipped with an Active Flutter Suppression System and Comparison with Wind-Tunnel Data. NASA TP-1367, Feb., 1979.

101. Edwards, J. W.: Unsteady Aerodynamic Modeling and Active Aeroelastic Control. Stanford University Report, SUDAAR - 504, Feb. 1977.

102. Nissim, E.: Flutter Suppression Using Active Controls Based on the Concept of Aerodynamic Energy. NASA TN D-6199, 1971.

103. Gangsaas, D. and Ly, U.: Application of a Modified Linear Quadratic Gaussion Design to Active Control of a Transport Airplane. AIAA Paper 79-1746, Aug., 1979.

104. Mukhopadhyay, V.; Newsom, J. R.; and Abel, I.: A Method for Obtaining Reduced Order Control Laws for High Order Systems Using Optimization Techniques. NASA TP 1876, 1981.

ASSUMPTIONS	EQUATION TYPE	FLOW REGIME	AEROELASTIC APPLICATIONS	EXAMPLE CODES
GENERAL VISCOUS COMPRESSIBLE FLUID CONTINUUM	NAVIER-STOKES	VISCOUS FLOW, TURBULENCE, SHOCK/BND.-LYR. INTER. SEPARATED FLOW	AILERON BUZZ, BUFFET, DYNAMIC STALL, STALL FLUTTER	STEGER & BAILEY
NO VISCOSITY	EULER	ATTACHED FLOWS, ROTATIONAL FLOW, CURVED SHOCKS	FLUTTER, ETC. WITH STRONG OR DETACHED SHOCKS, BOUNDARY-LAYER CONVECTION.	MAGNUS AND YOSHIHARA,
IRROTATIONAL, ISENTROPIC	FULL POTENTIAL	MODERATE FLOW PERTURBATIONS, WEAK SHOCKS, LARGE SHOCK MOTIONS	FLUTTER, ETC. WITH FINITE THICKNESS AND INCIDENCE.	ISOGAI GOORJIAN MALONE SHANKAR KANDIL & YATES
SMALL THICKNESS, INCIDENCE, AND AMPLITUDE	TRANSONIC SMALL PERTURBATION (TSP)	SMALL FLOW PERTURBATIONS, WEAK SHOCKS MODERATE SHOCK MOTIONS	NONLINEAR AIRLOADS	LTRAN2 SUSAN XTRAN2L XTRAN3S
AMPLITUDE << THICKNESS AND INCIDENCE	LINEARIZED TSP (LTSP)	LIMITED SHOCK MOTION	LINEAR AIRLOADS, SMALL MOTIONS OF THIN LIFTING SURFACES	OPTRAN2 OPTRAN3
MACH NUMBER NOT NEAR UNITY	CLASSICAL LINEAR THEORY	SUBSONIC AND SUPERSONIC	FLUTTER, ETC.	KERNEL FUNCTION DOUBLET LATTICE RHOIV, MACH BOX SOUSSA/UTSA

Table 1. Hierarchy of governing equations for unsteady aerodynamics.

M RANGE / α RANGE	SUBSONIC	TRANSONIC	SUPERSONIC
LOW (ATTACHED FLOW) W/LARGE CONTROL DEFLECTION	UTSA SVP	NONLINEAR UTSA SVP	UTSA
MODERATE (VORTEX SEPARATION) W/LARGE CONTROL DEFLECTION	UTSA+ HYBRID VORTEX SVP	NONLINEAR UTSA+ HYBRID VORTEX SVP	
LARGE (SEPARATED FLOW) W/ OR W/O CONTROL DEFLECTION	SVP SVP	SVP SVP	

Table 2. Summary of integral-equation activities.

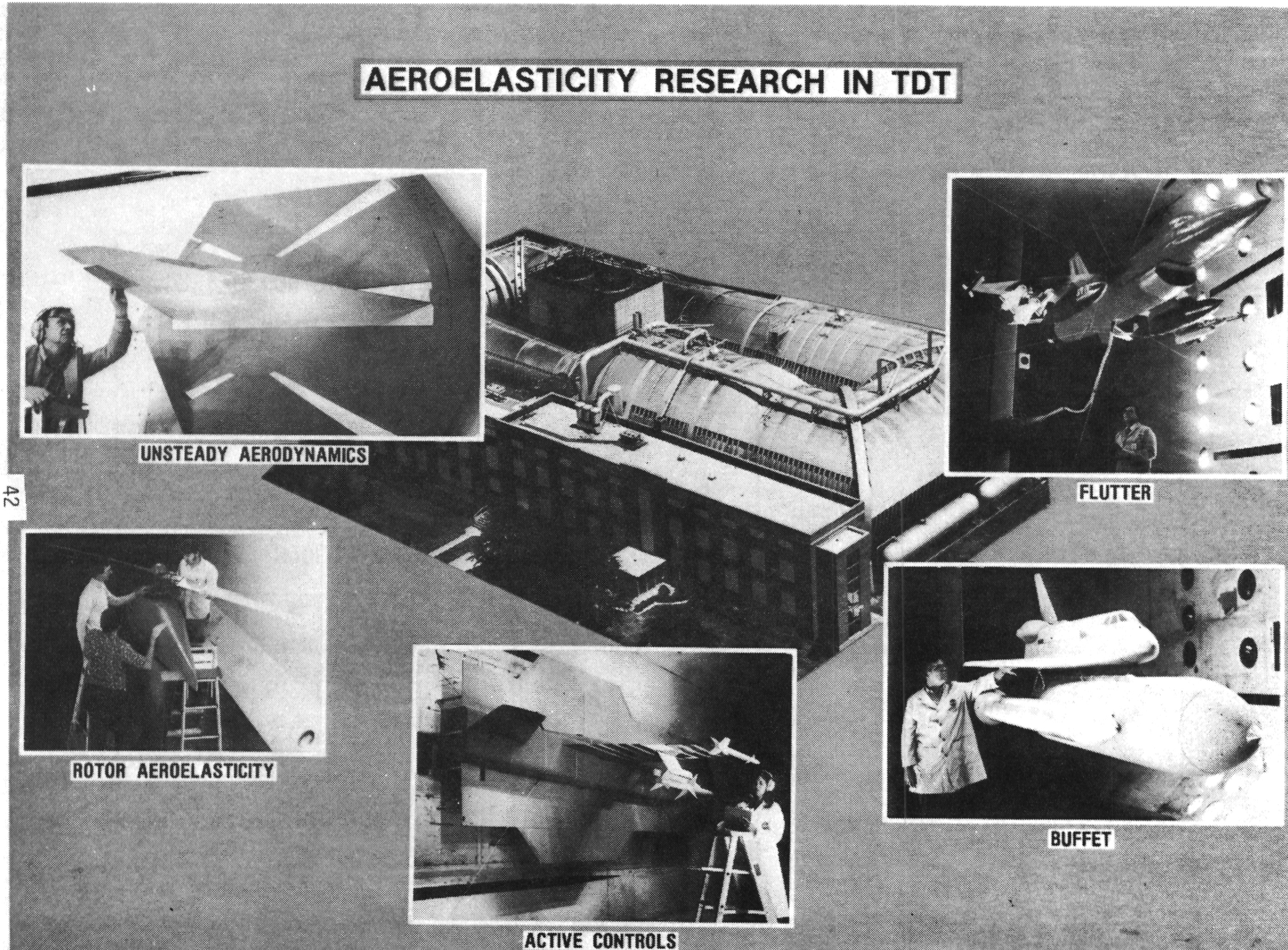

Figure 1. Aeroelastic technology areas supported by the Langley Research Center Transonic Dynamics Tunnel (TDT).

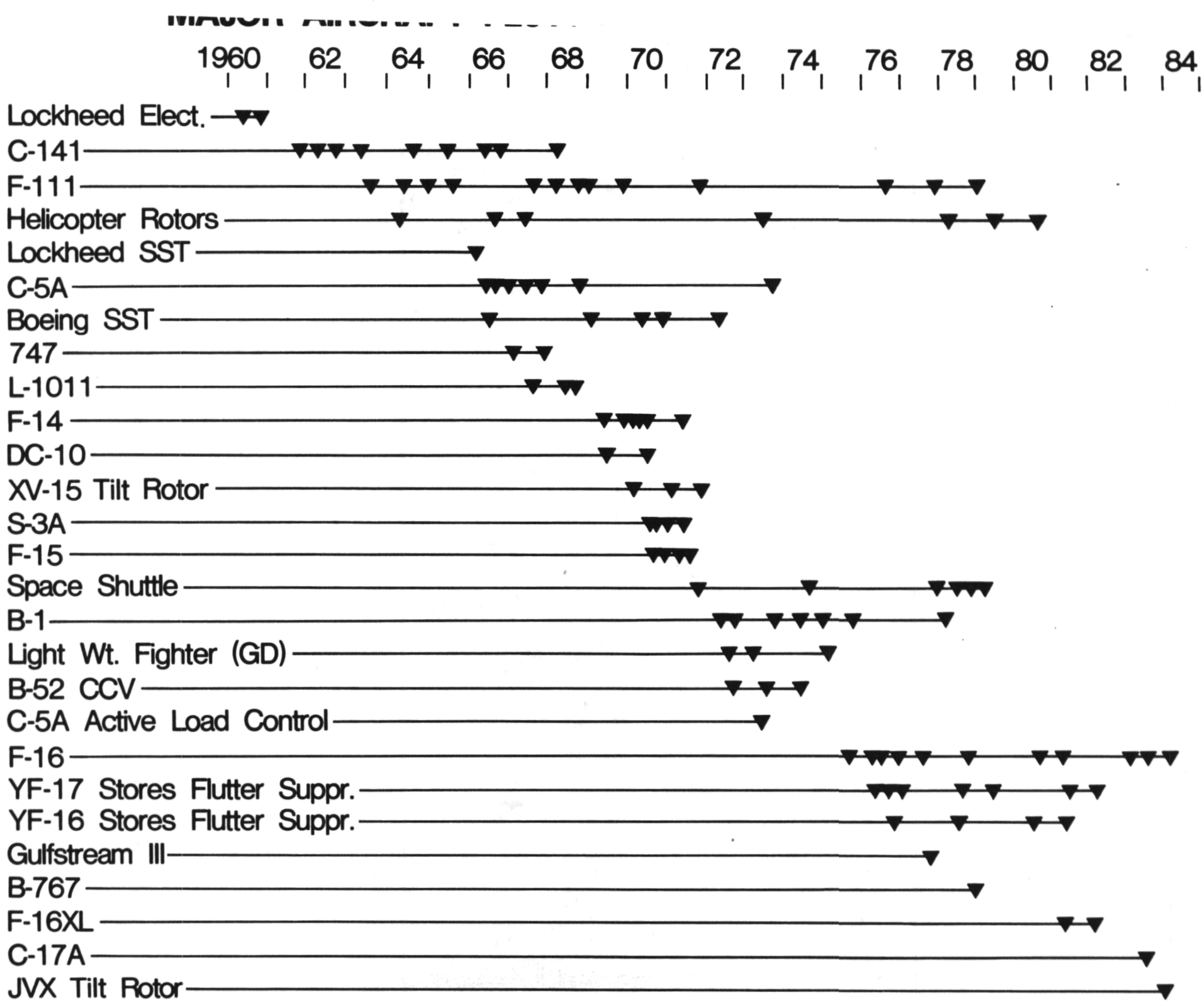

Figure 2. Some of the major aircraft flutter investigations in the TDT.

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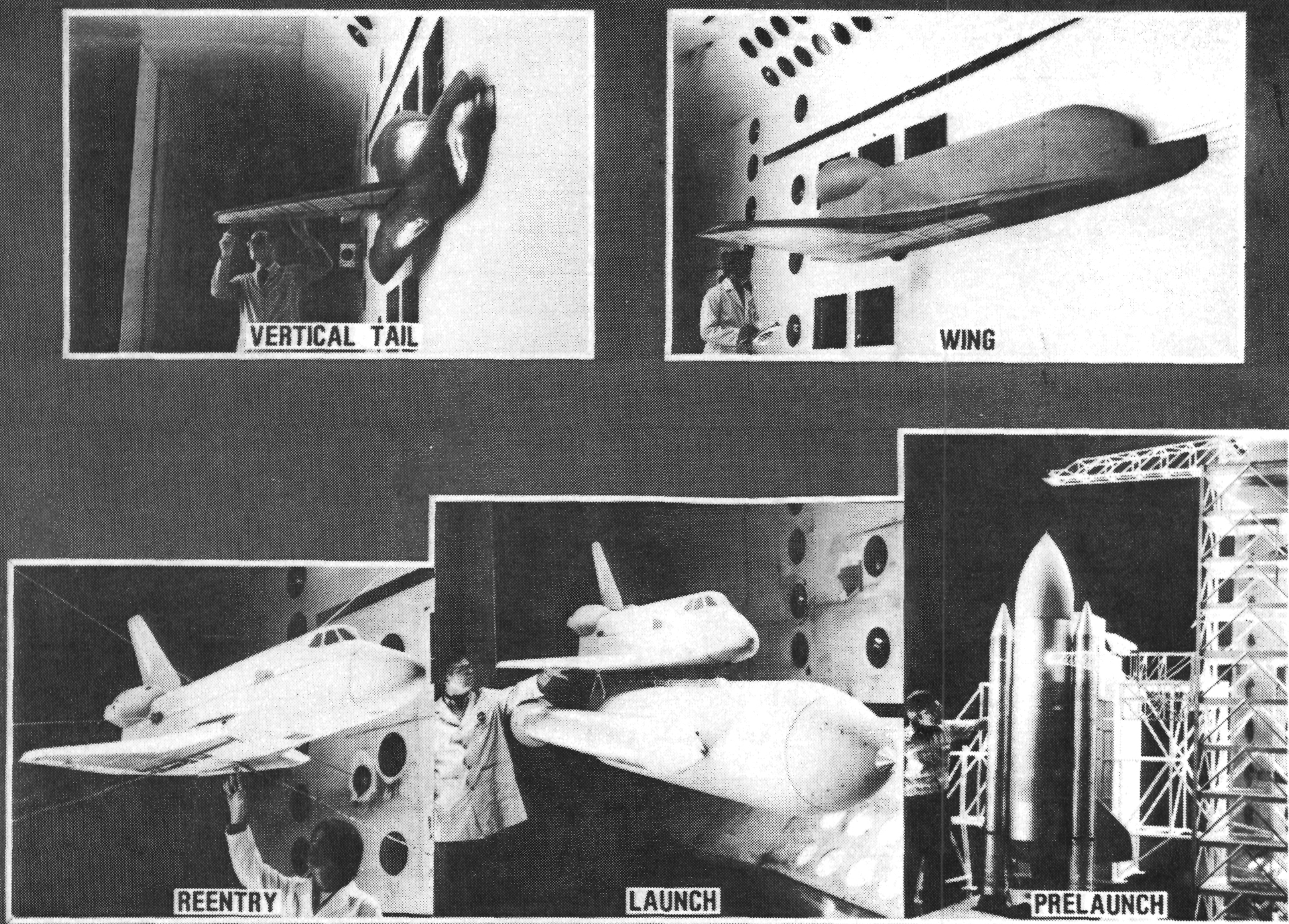


Figure 3. Space Shuttle aeroelastic model studies in the TDT.

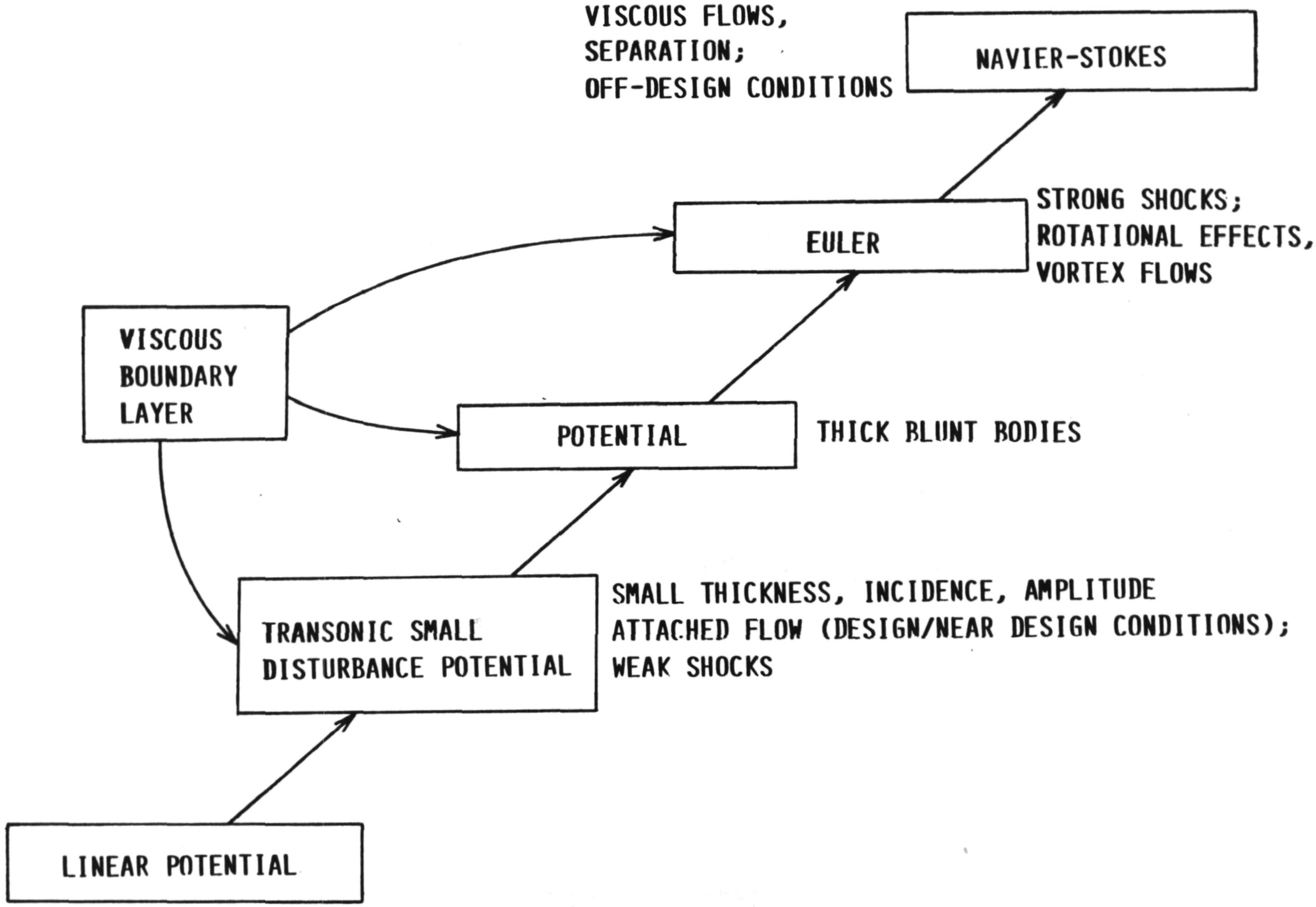


Figure 4. Progression up the hierarchy of flow equations.

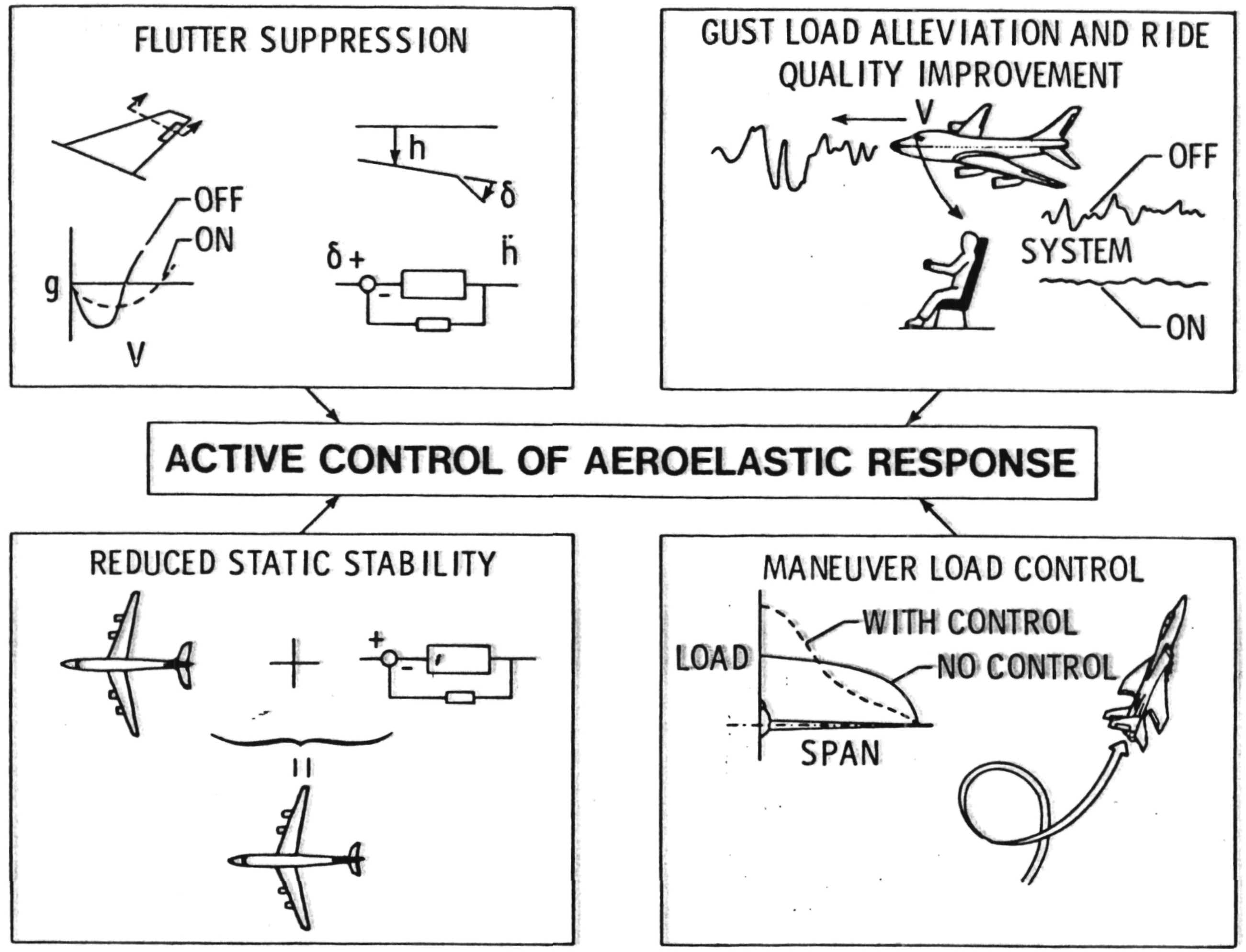


Figure 5. Active control functions associated with the control of aeroelastic stability and response.

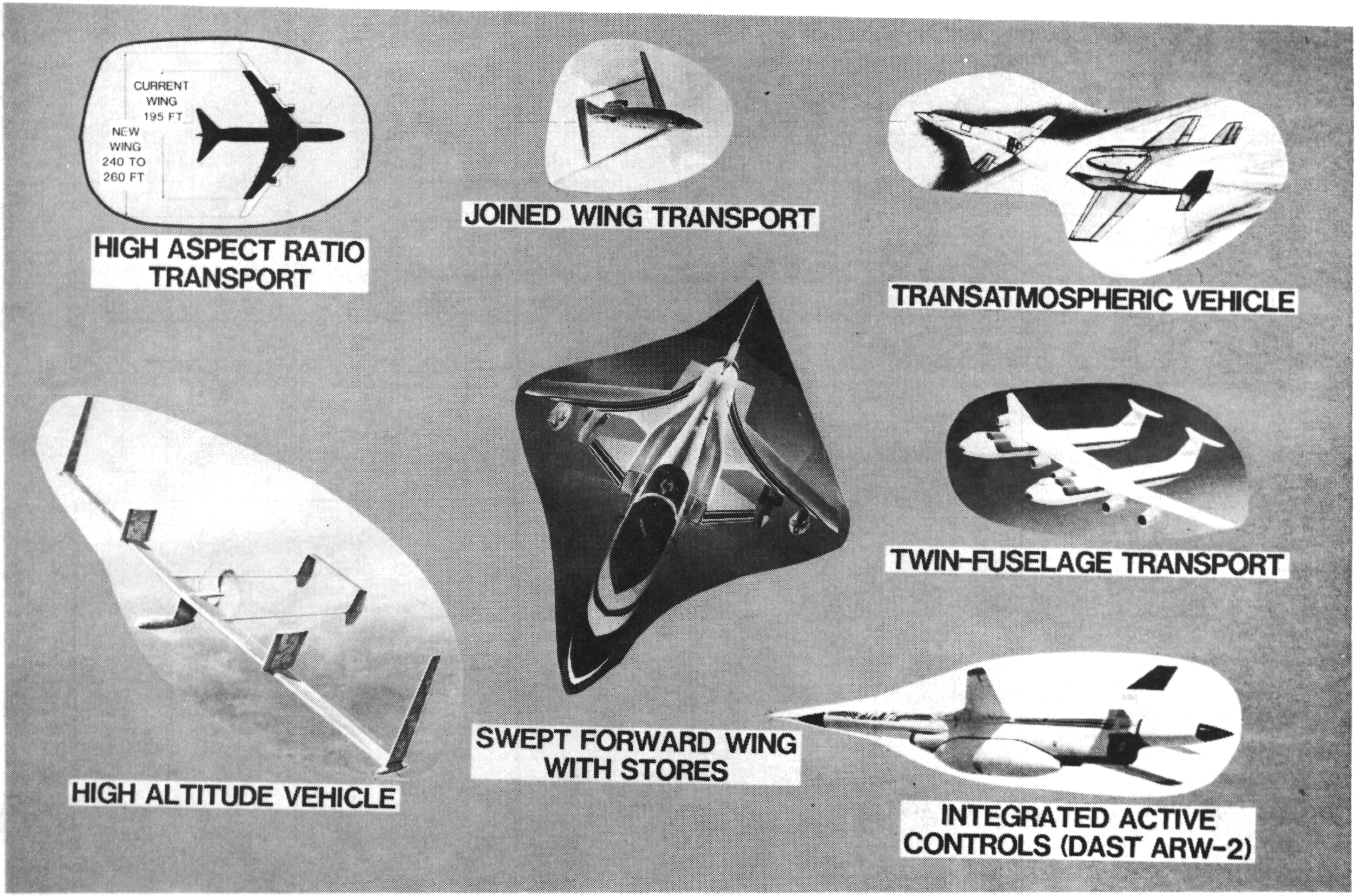
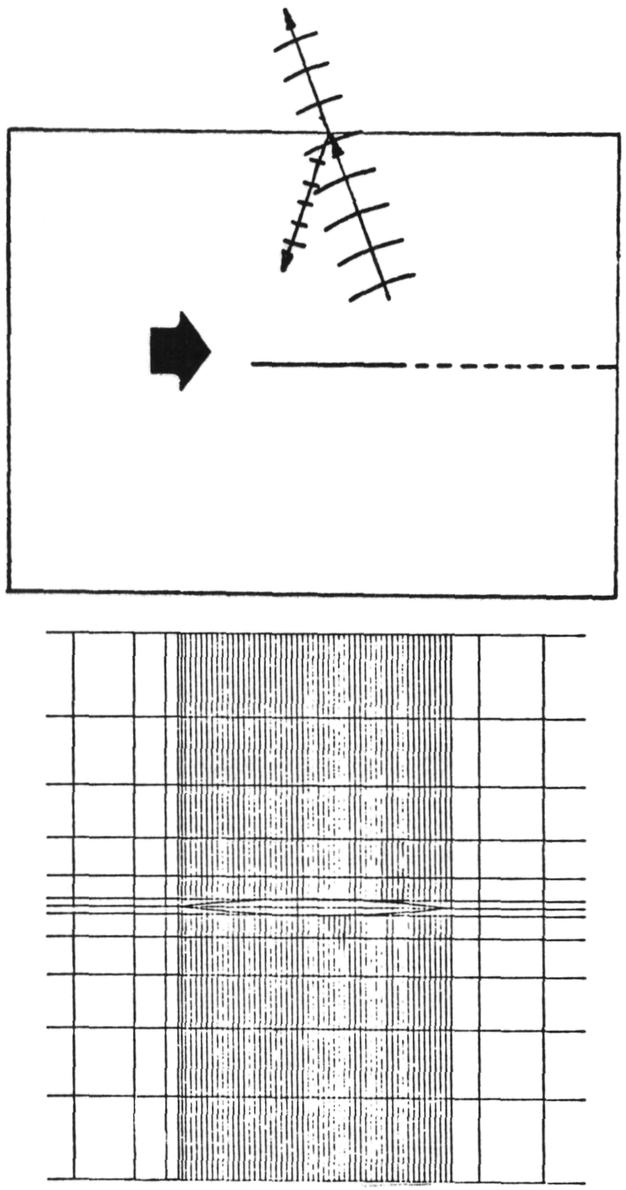


Figure 6. - Examples of innovative designs that can pose new challenges in aeroelasticity.



- MODIFIED GRID
- NONREFLECTING BOUNDARY CONDITIONS
- MORE ACCURATE
- 33% SAVINGS IN COMPUTER COSTS

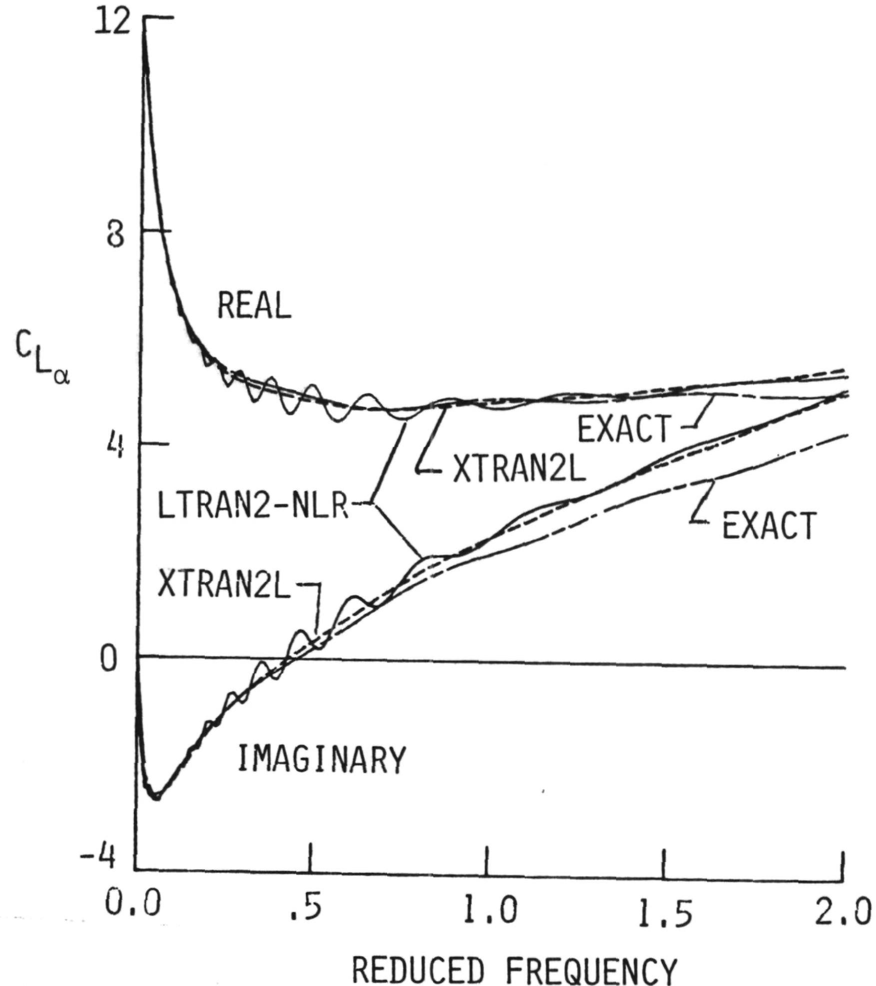


Figure 7. Improved modeling increases accuracy of unsteady transonic aerodynamics calculations.

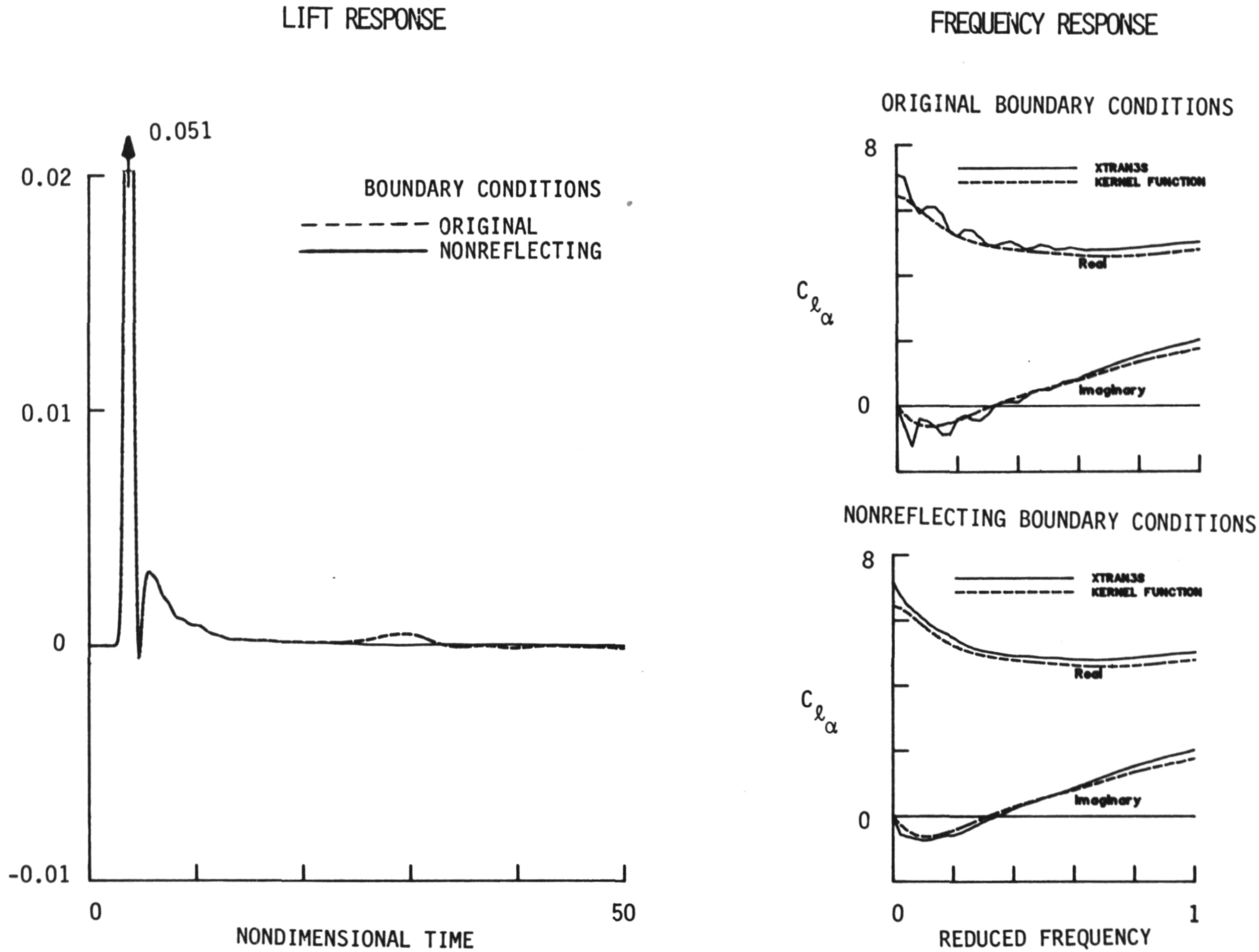


Figure 8. Non-reflective far-field boundary conditions improve unsteady airload calculations.

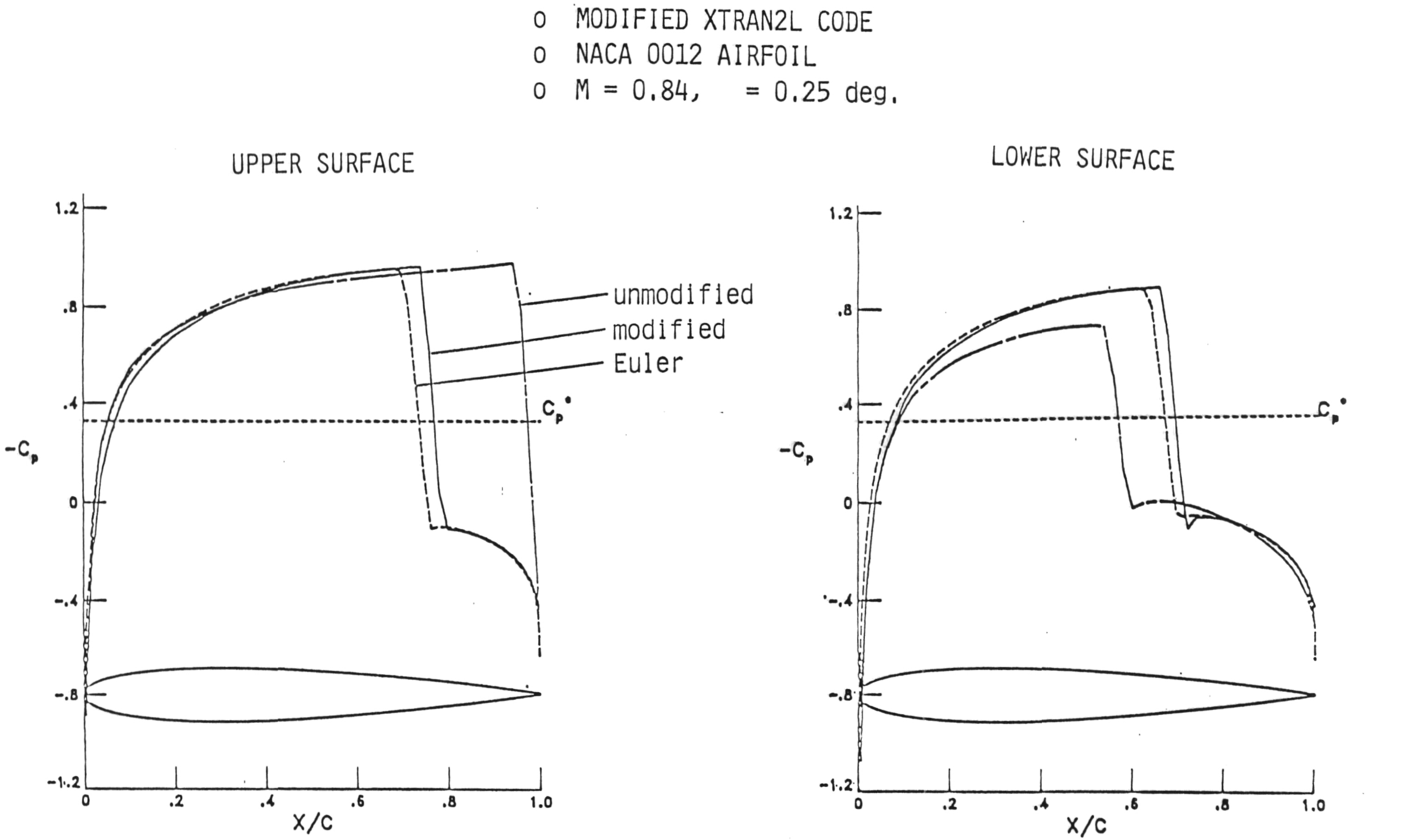


Figure 9. Non-isentropic small disturbance theory resolves non-uniqueness problems.

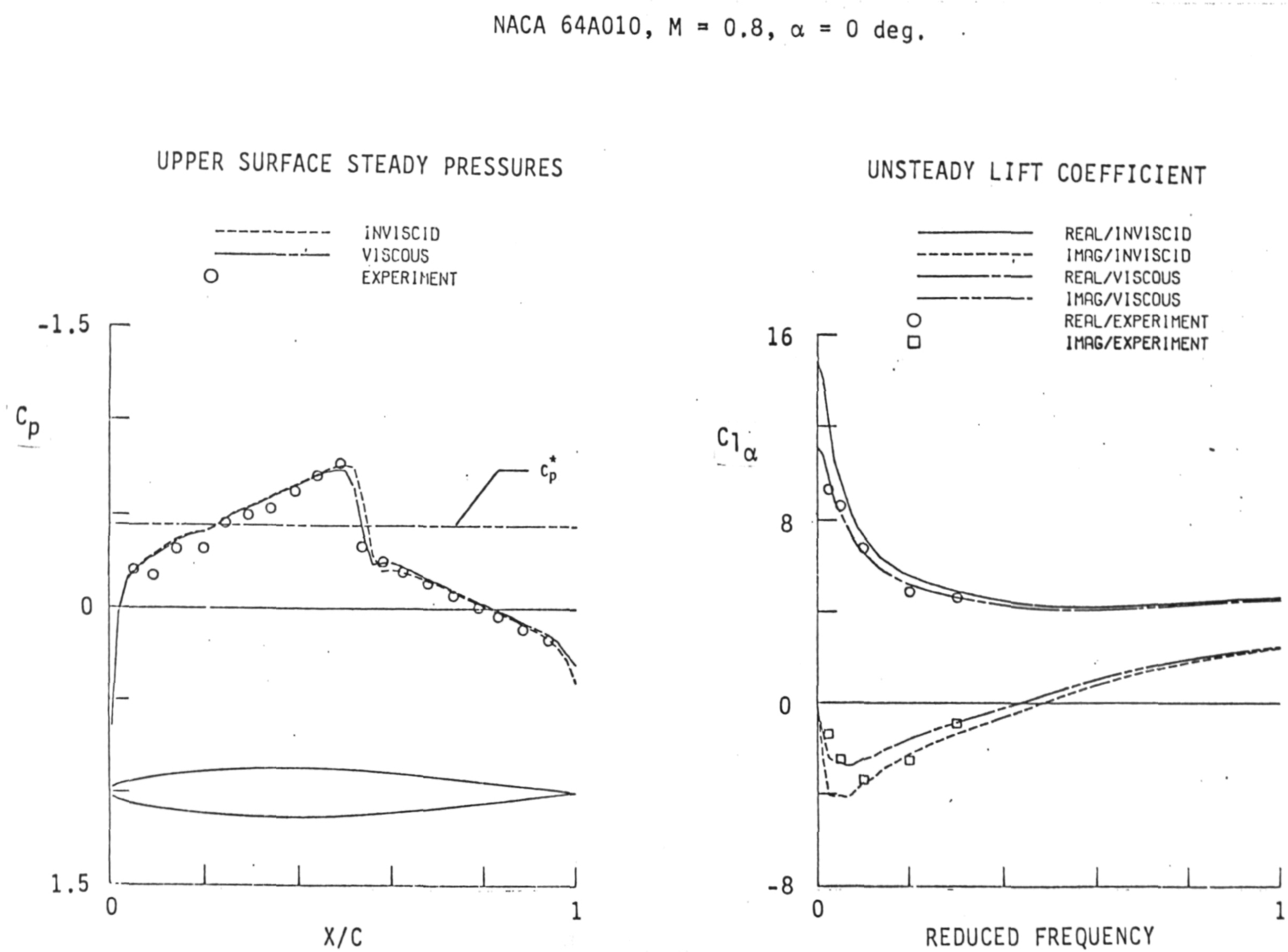
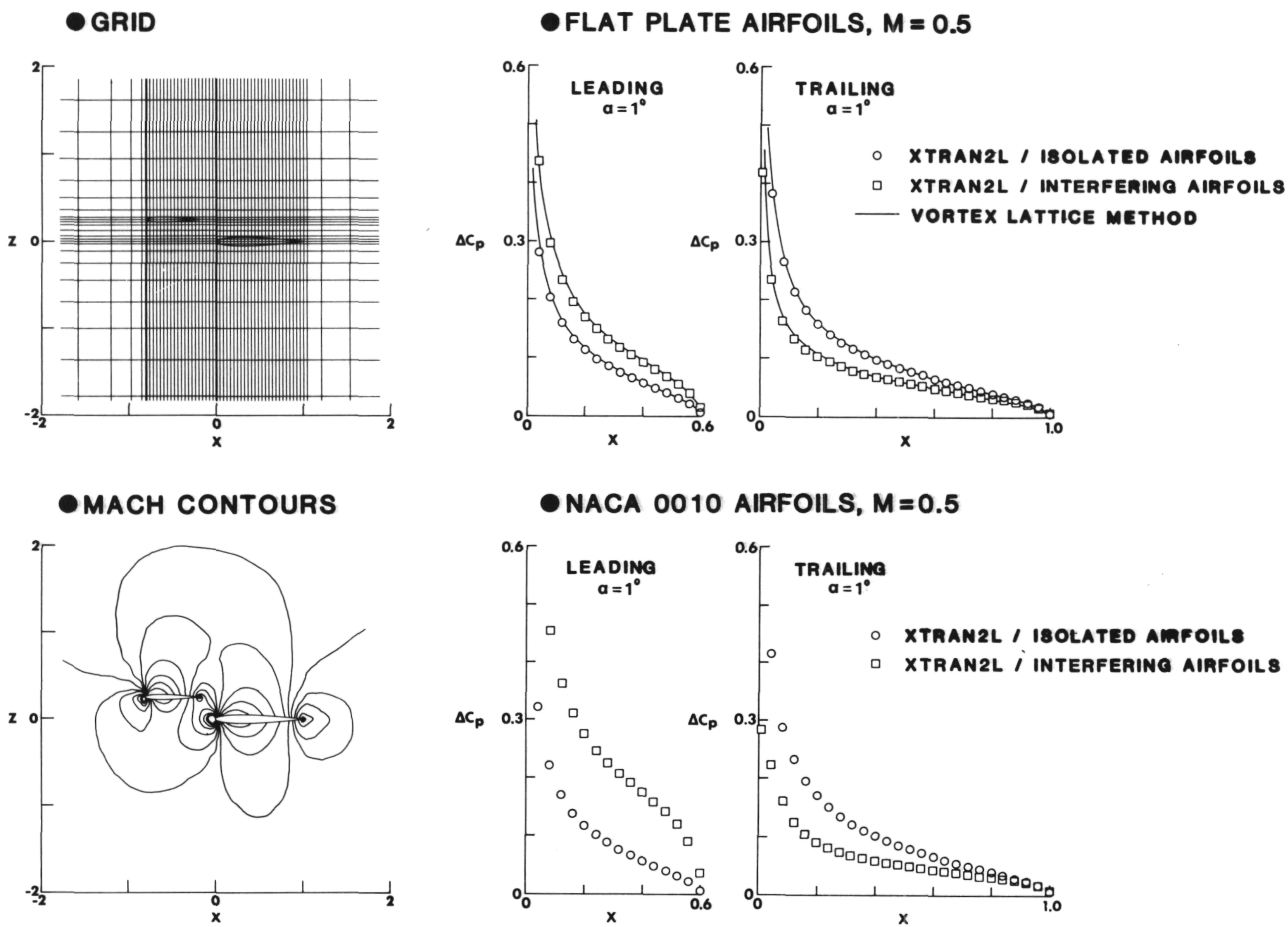


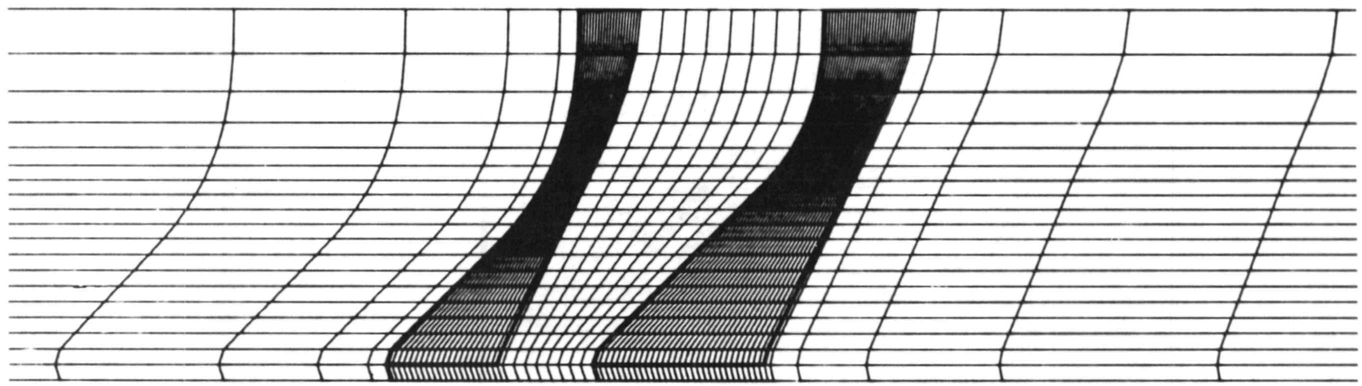
Figure 10. Efficient, accurate viscous boundary layer model coupled with XTRAN2L code.



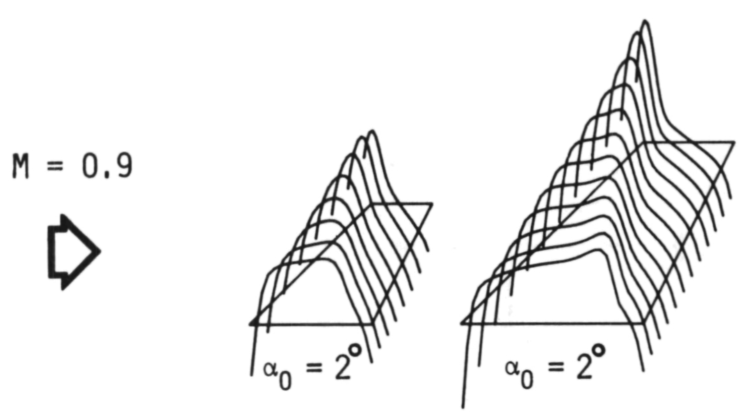
(a) XTRAN2L

Figure 11. XTRAN multiple lifting surface capability.

● Canard-wing near-field grid



● Transonic steady pressure distributions on canard and wing upper surfaces



(b) XTRAN3S

Figure 11. Concluded.

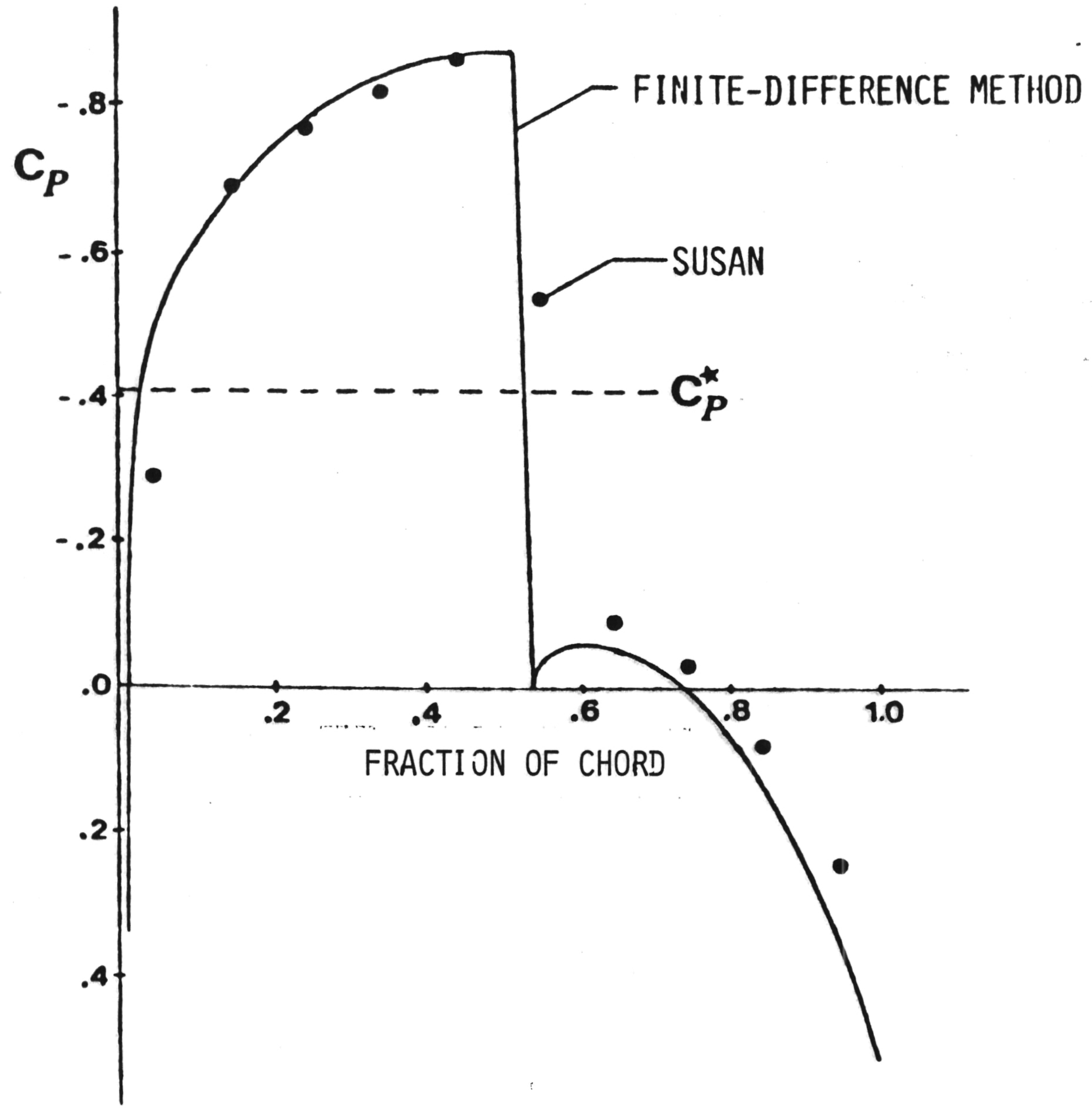


Figure 12. - Pressure distribution near root of an aspect-ratio 6 rectangular wing with NACA 0012 airfoil at Mach number 0.82, α = 0.

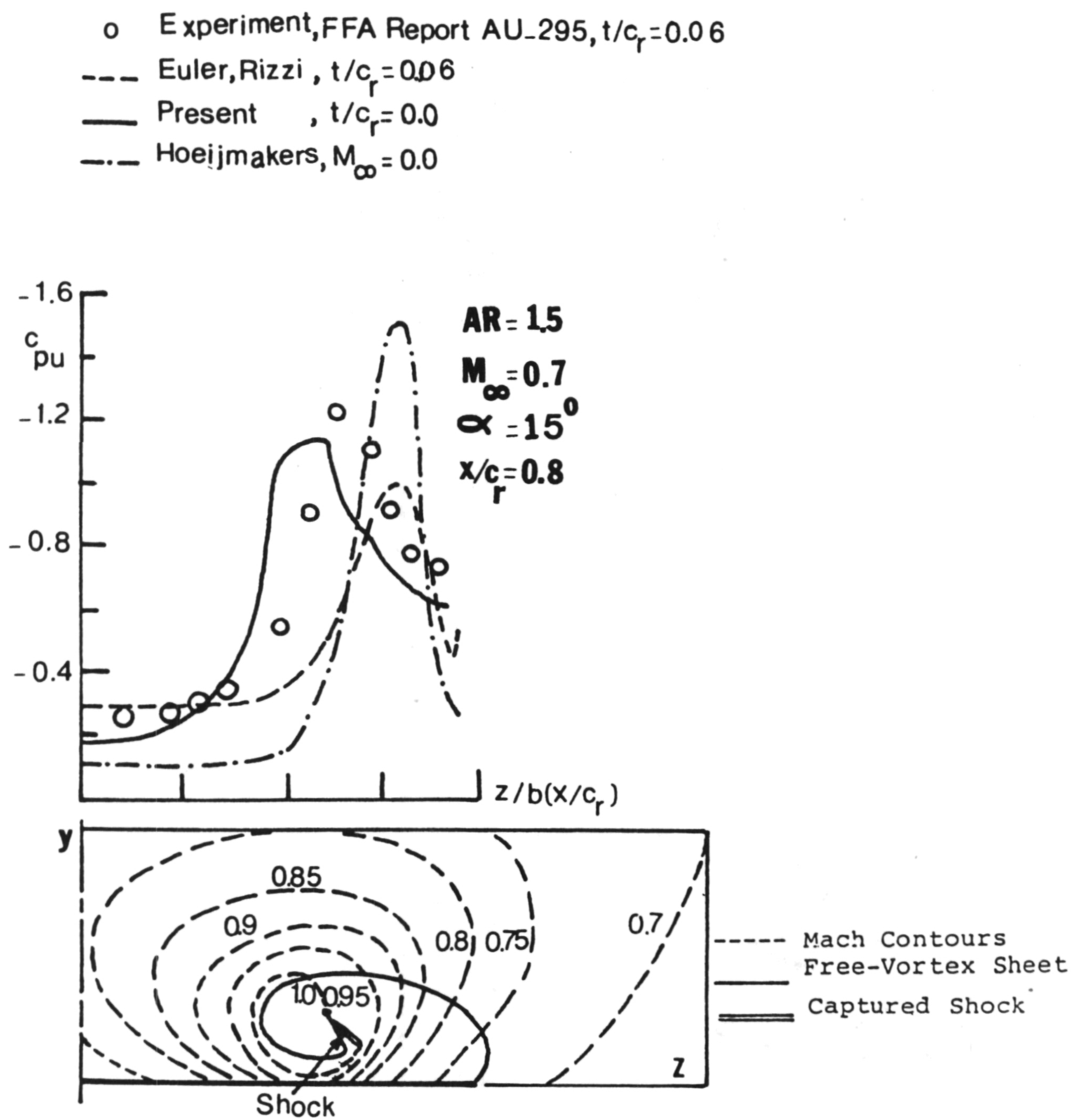


Figure 13. - Spanwise pressure variation and flow in cross-flow plane at 0.8 root chord of delta wing.

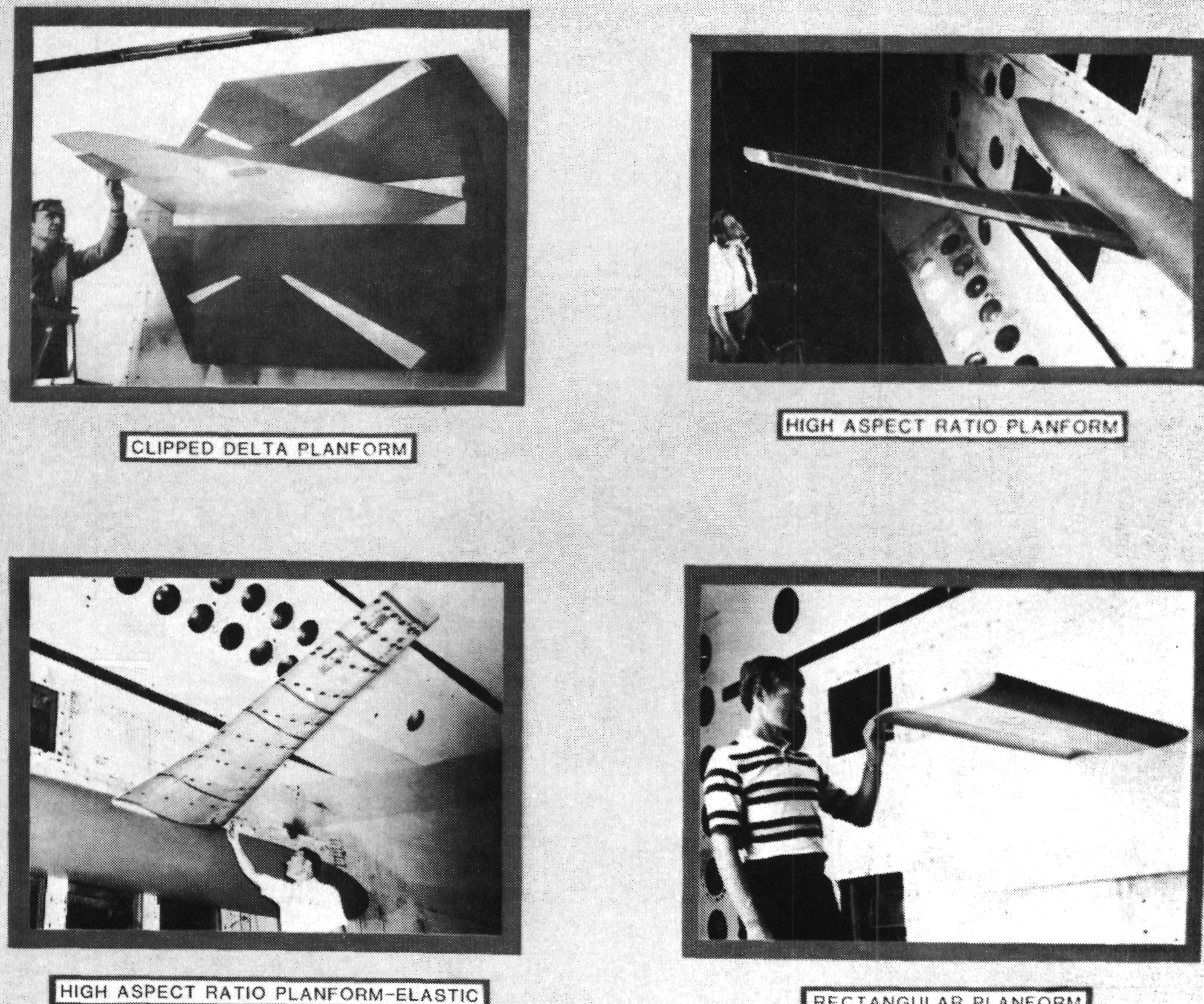


Figure 14. Unsteady pressure models mounted in the TDT.

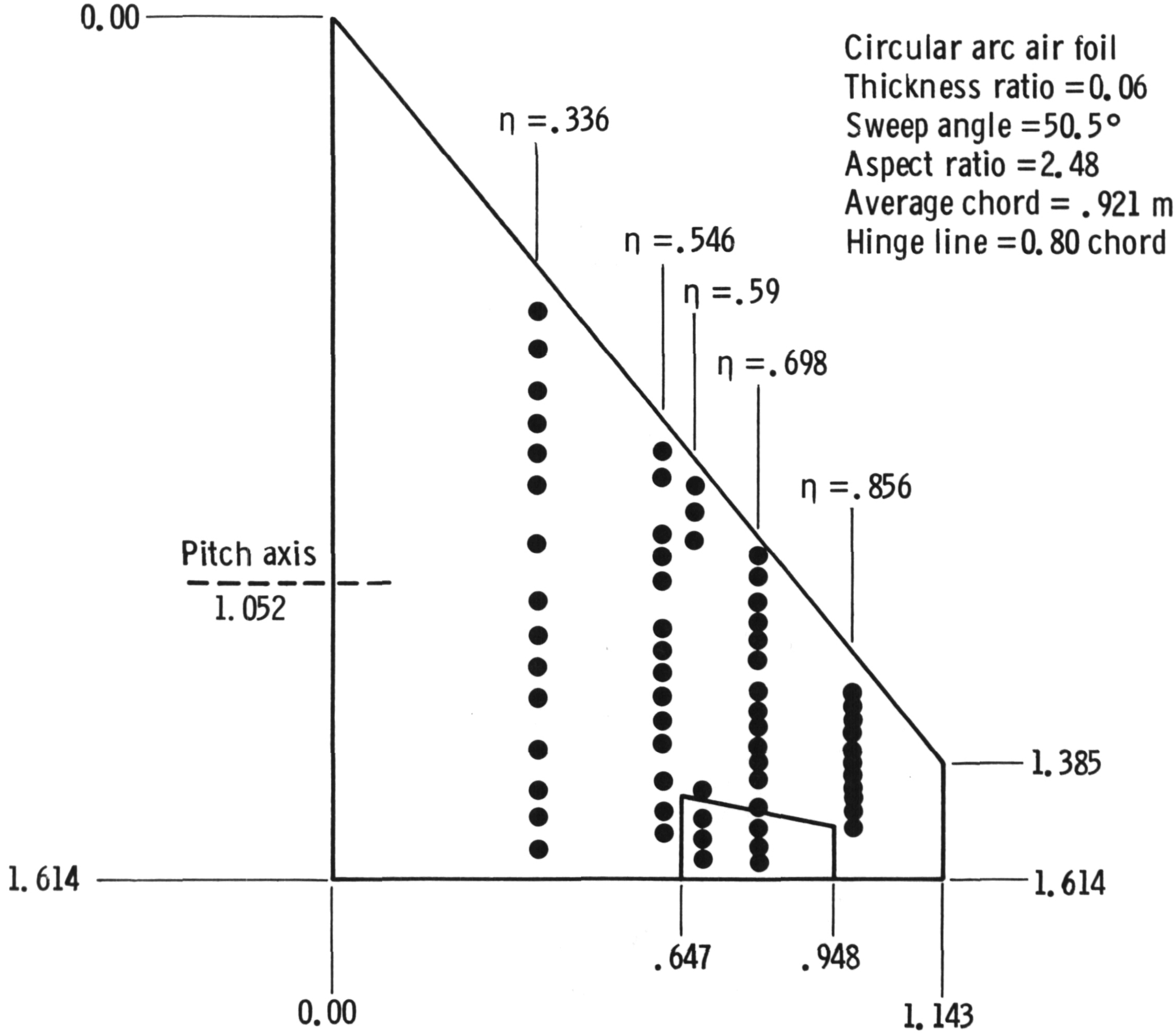


Figure 15. Sketch of clipped-tip delta wing unsteady pressure model.

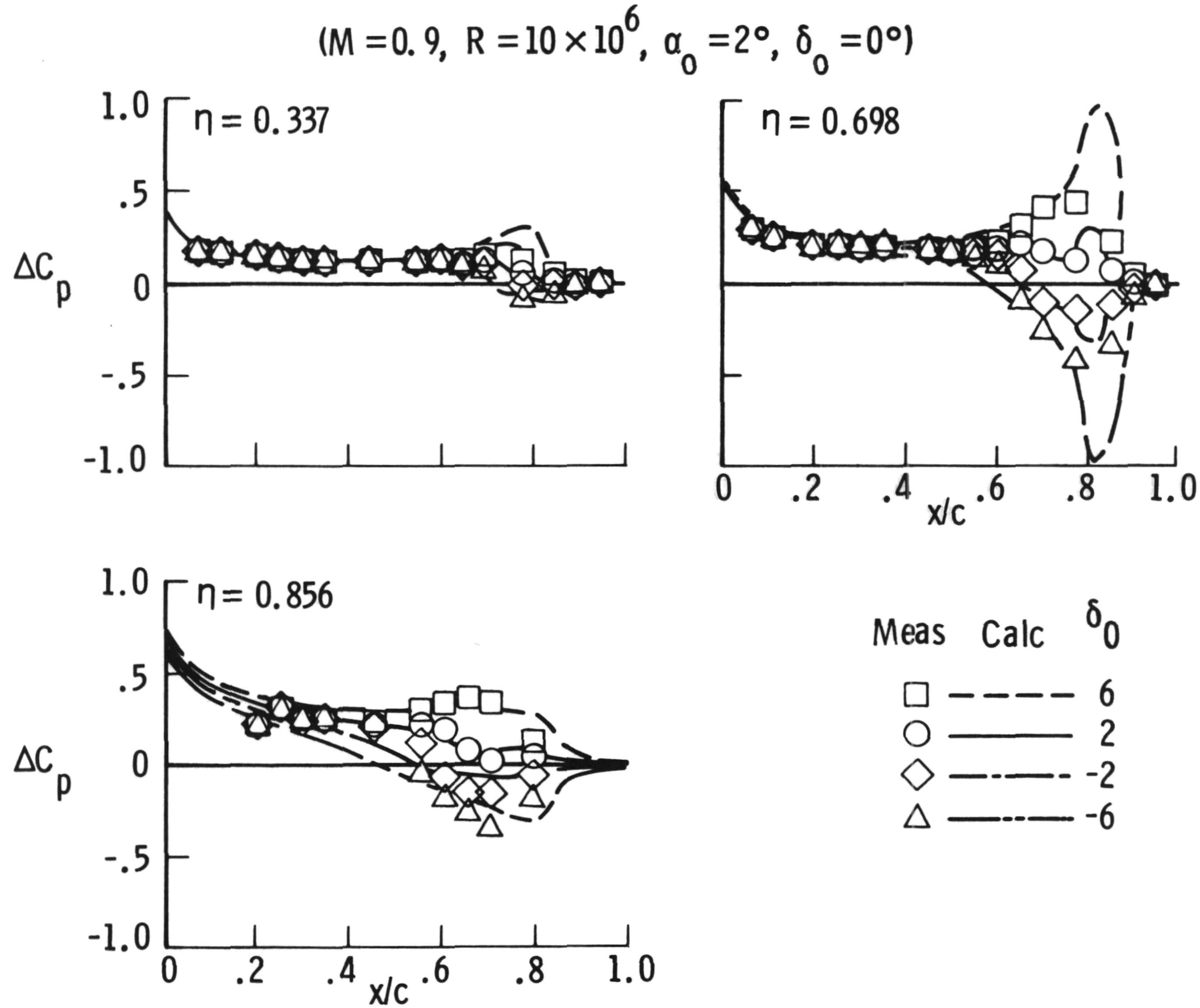


Figure 16. Steady lifting pressure distribution variation with control surface deflection at M = 0.90 for three span stations along delta wing model.

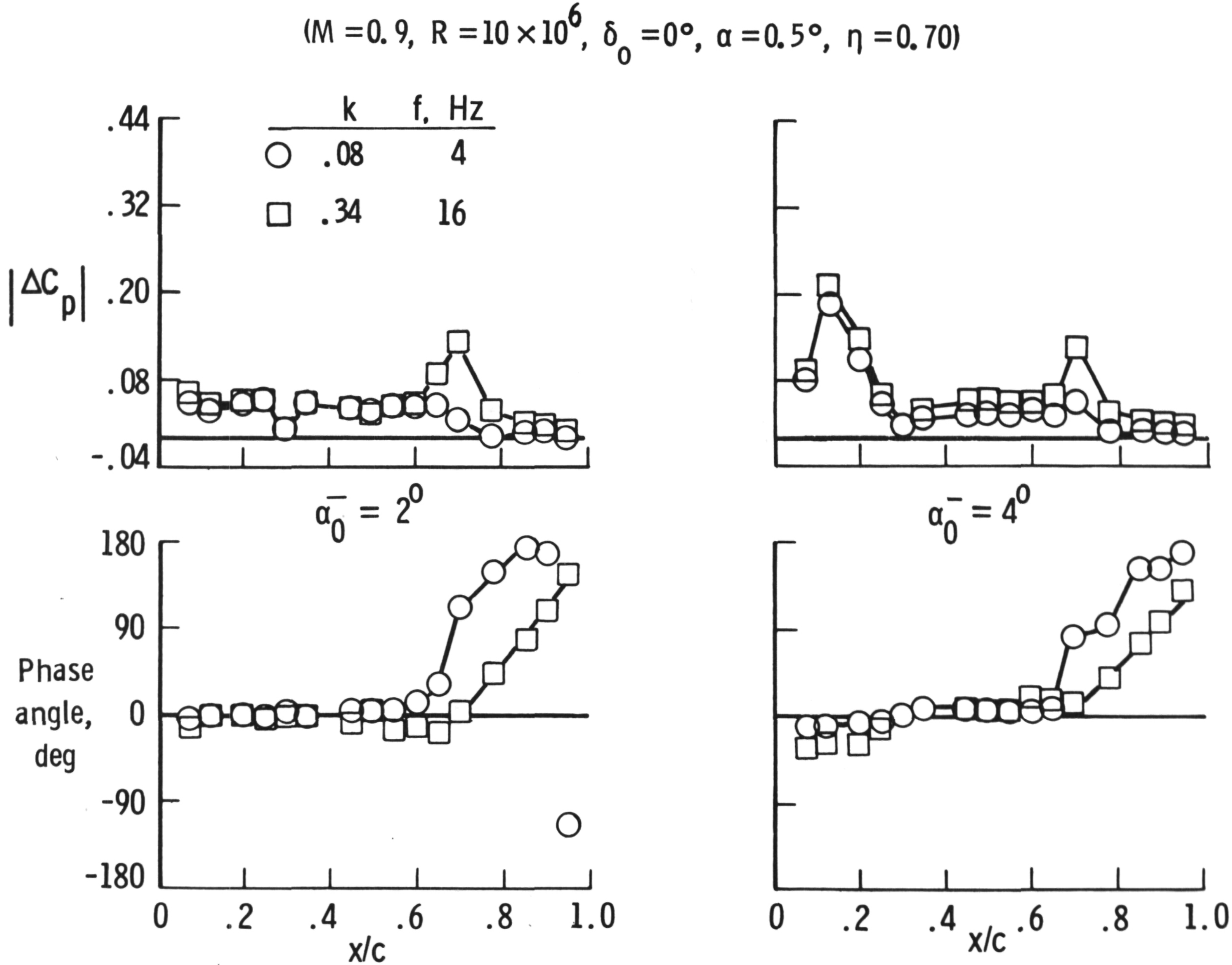


Figure 17. Delta wing unsteady lifting pressure magnitude and phase angle distribution variation with pitch oscillations of 0.5° at M = 0.90 at 70-percent-span.

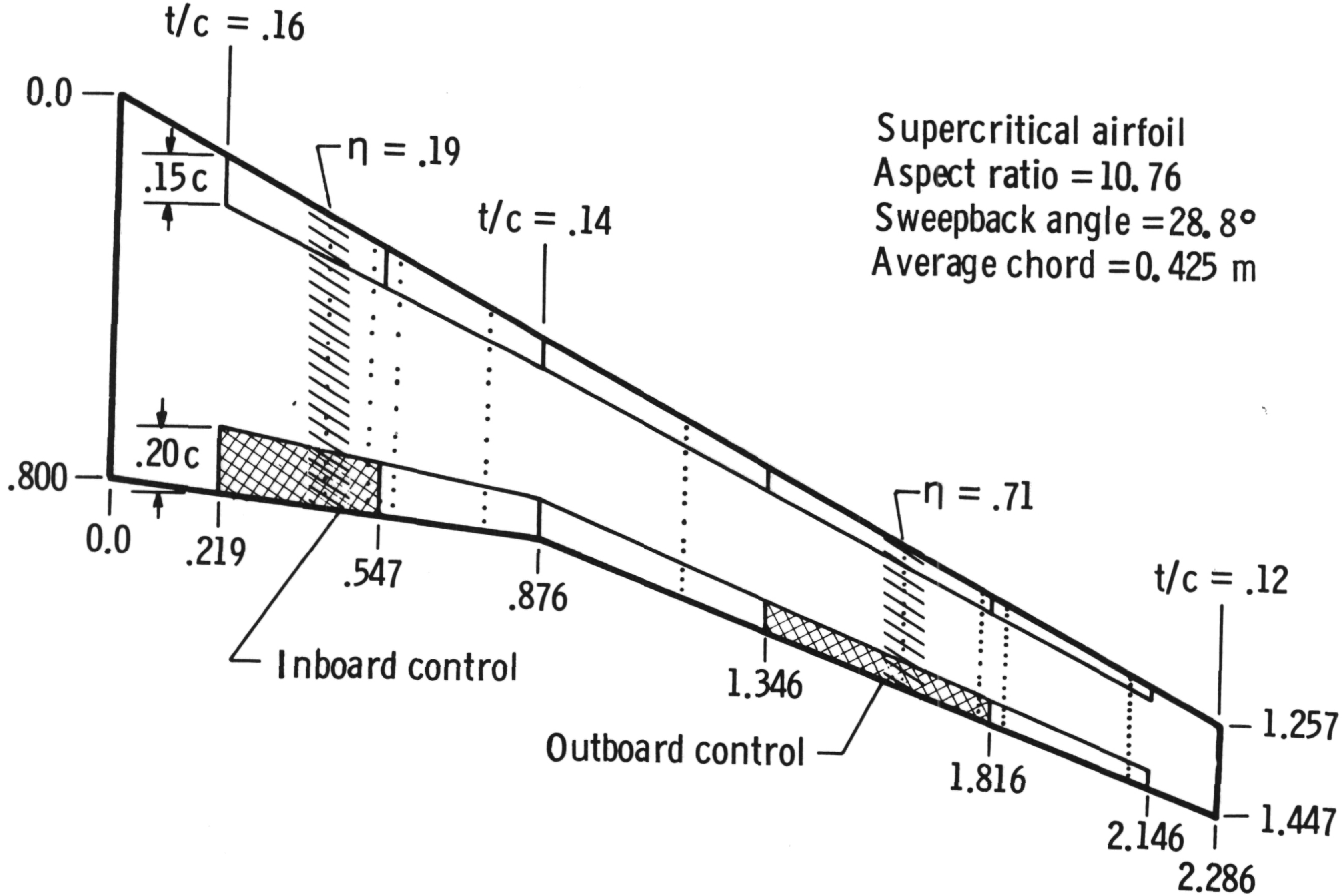


Figure 18. Sketch of high-aspect-ratio unsteady pressure model.

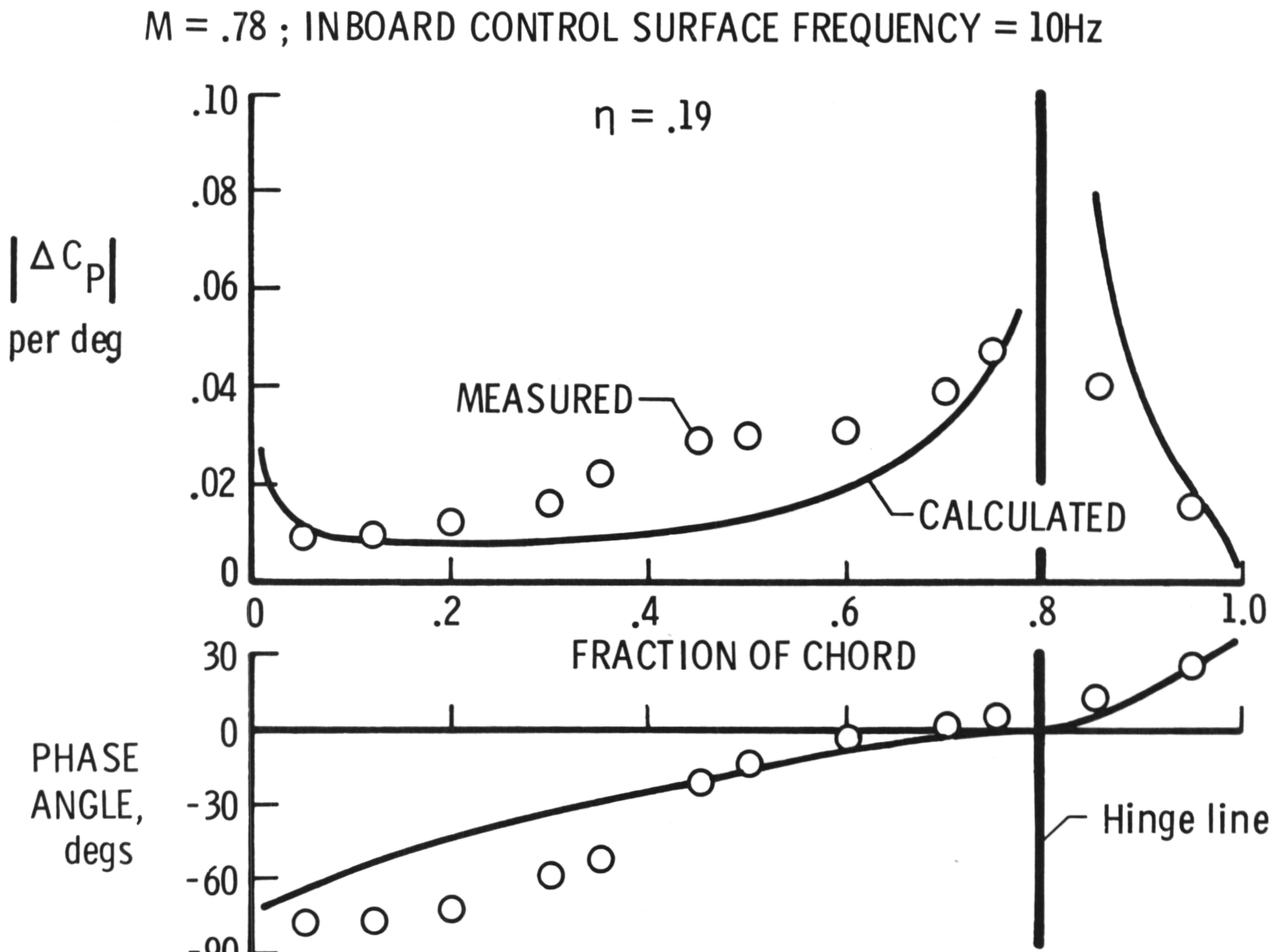


Figure 19. Comparison of measured and calculated (RHOIV) chordwise unsteady lifting pressure distribution for inboard control surface of high-aspect-ratio model.

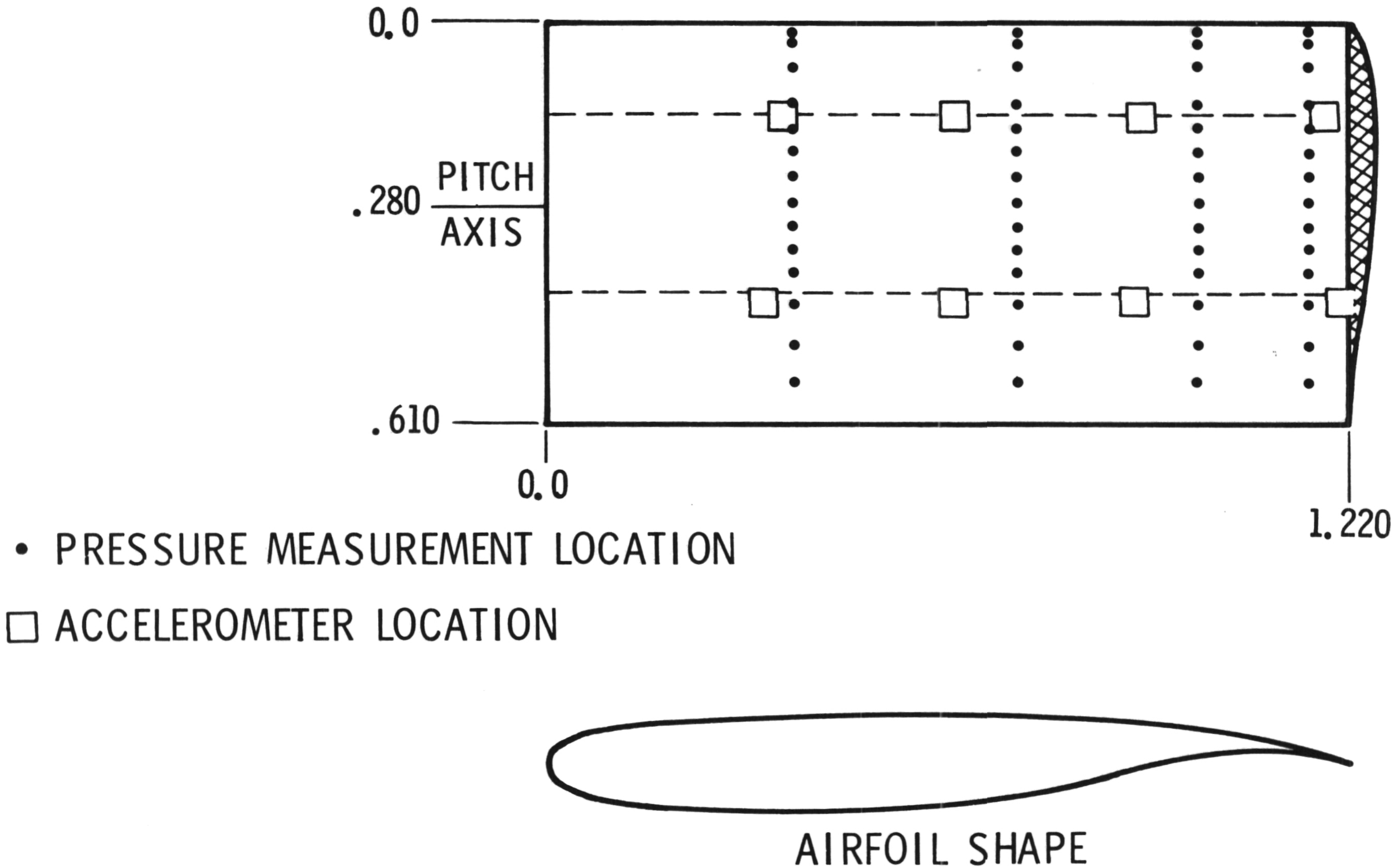


Figure 20. Sketch of planform and airfoil shape of rectangular supercritical wing model (dimensions in meters).

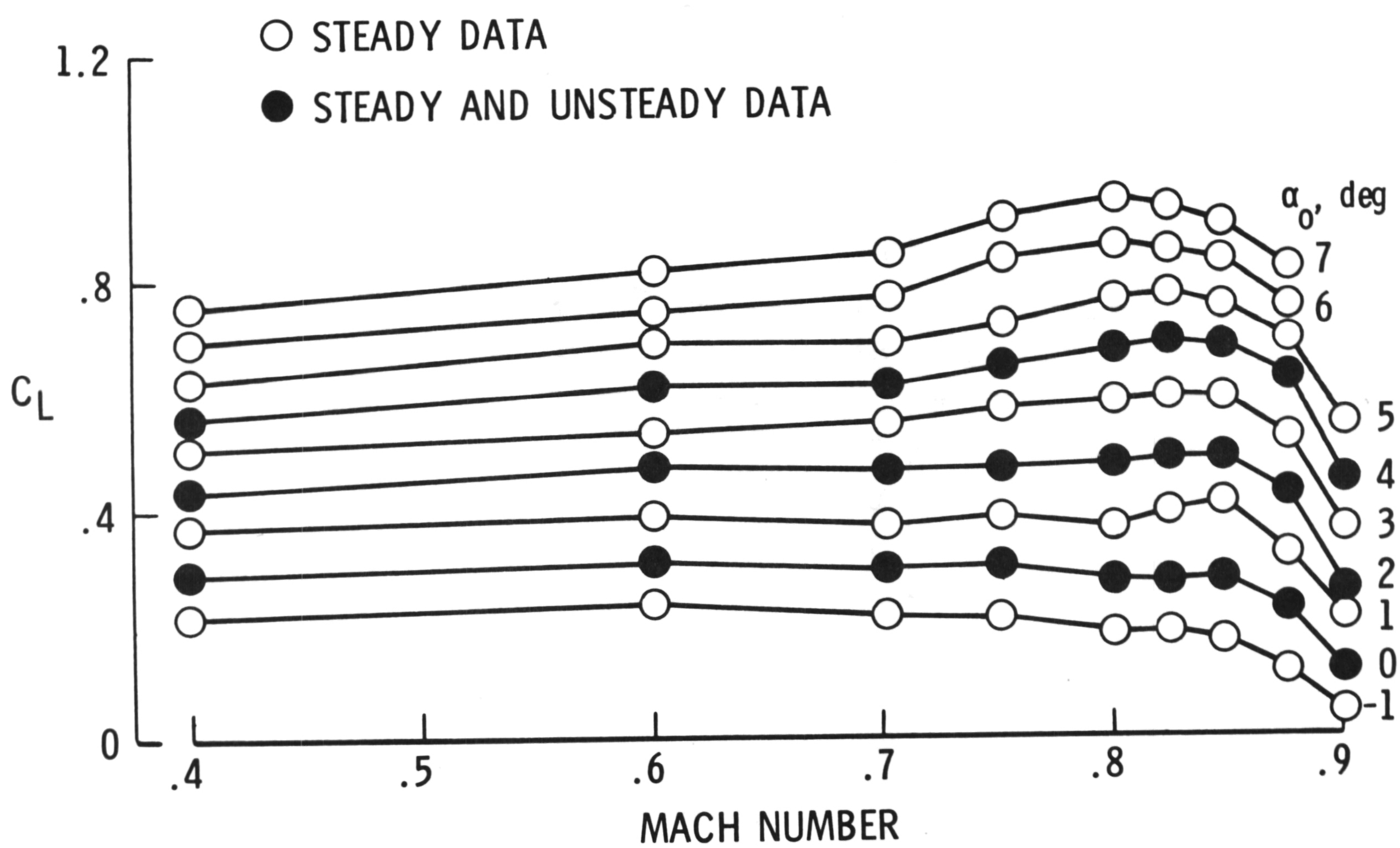


Figure 21. Variation of total lift coefficient with Mach number for rectangular supercritical wing for various angles of attack.

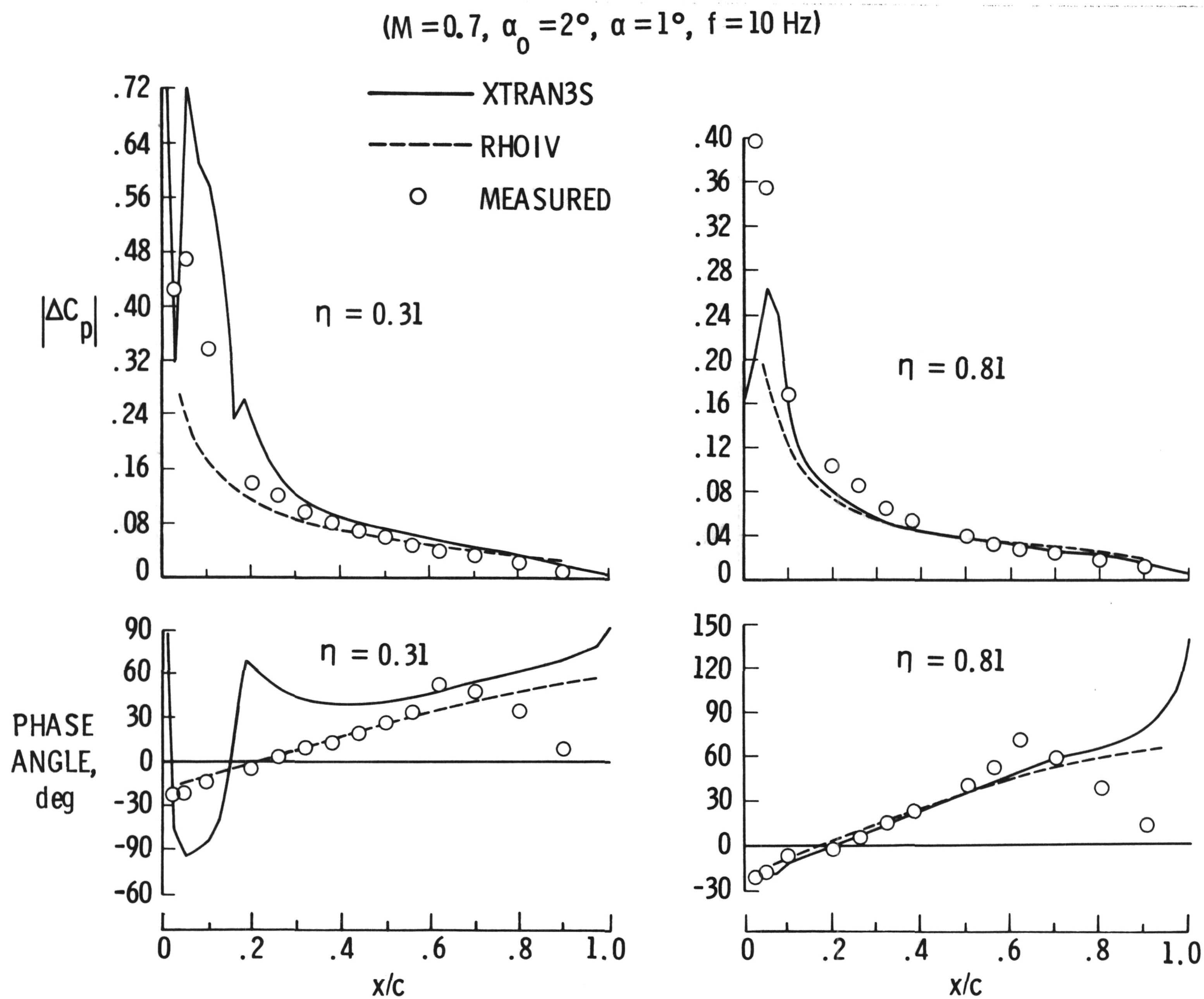


Figure 22. Comparison of measured and calculated unsteady pressure magnitude and phase distributions at two span stations of rectangular wing.

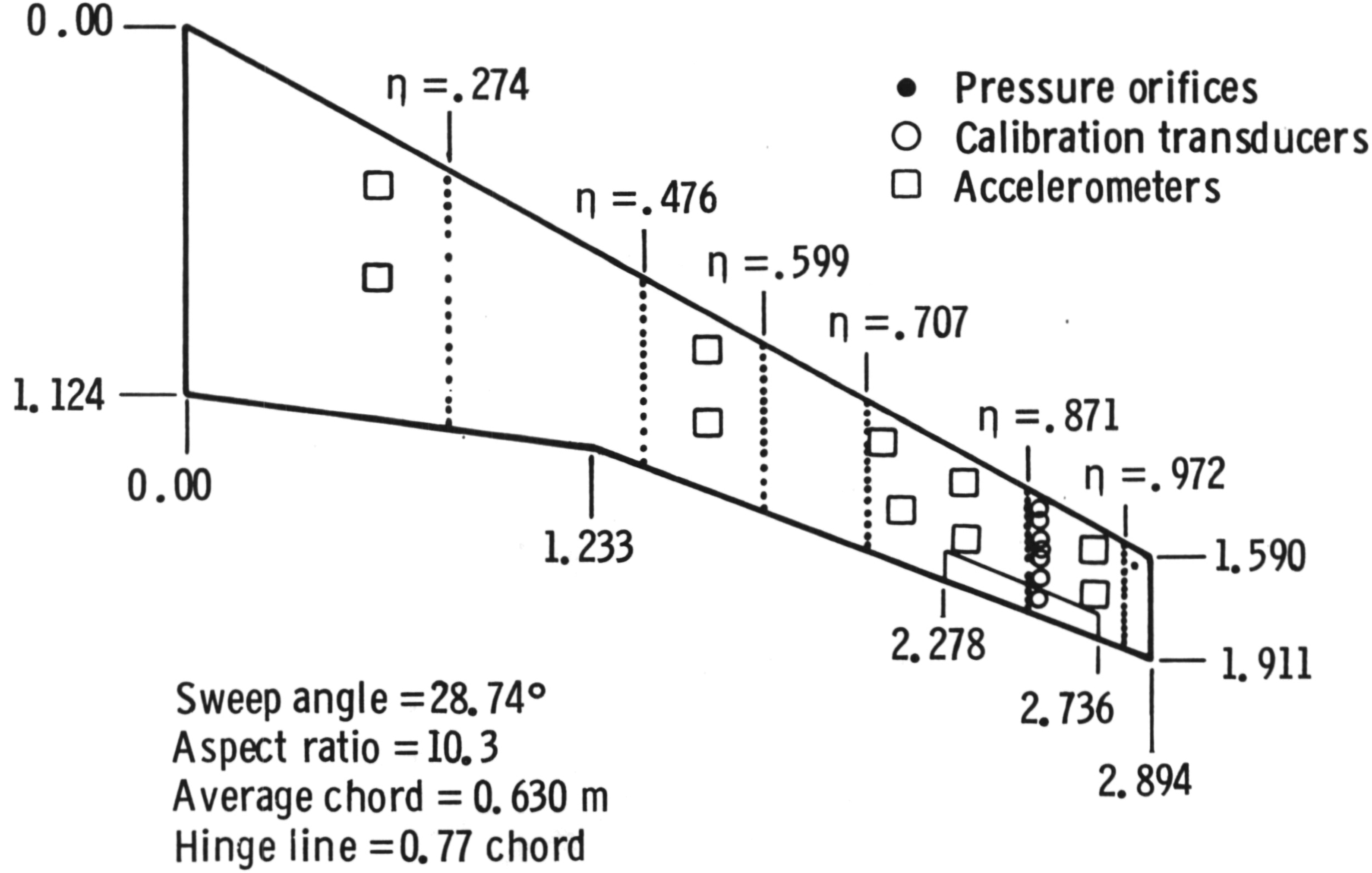


Figure 23. Sketch of planform and instrumenttion locations for a flexible unsteady pressure model.

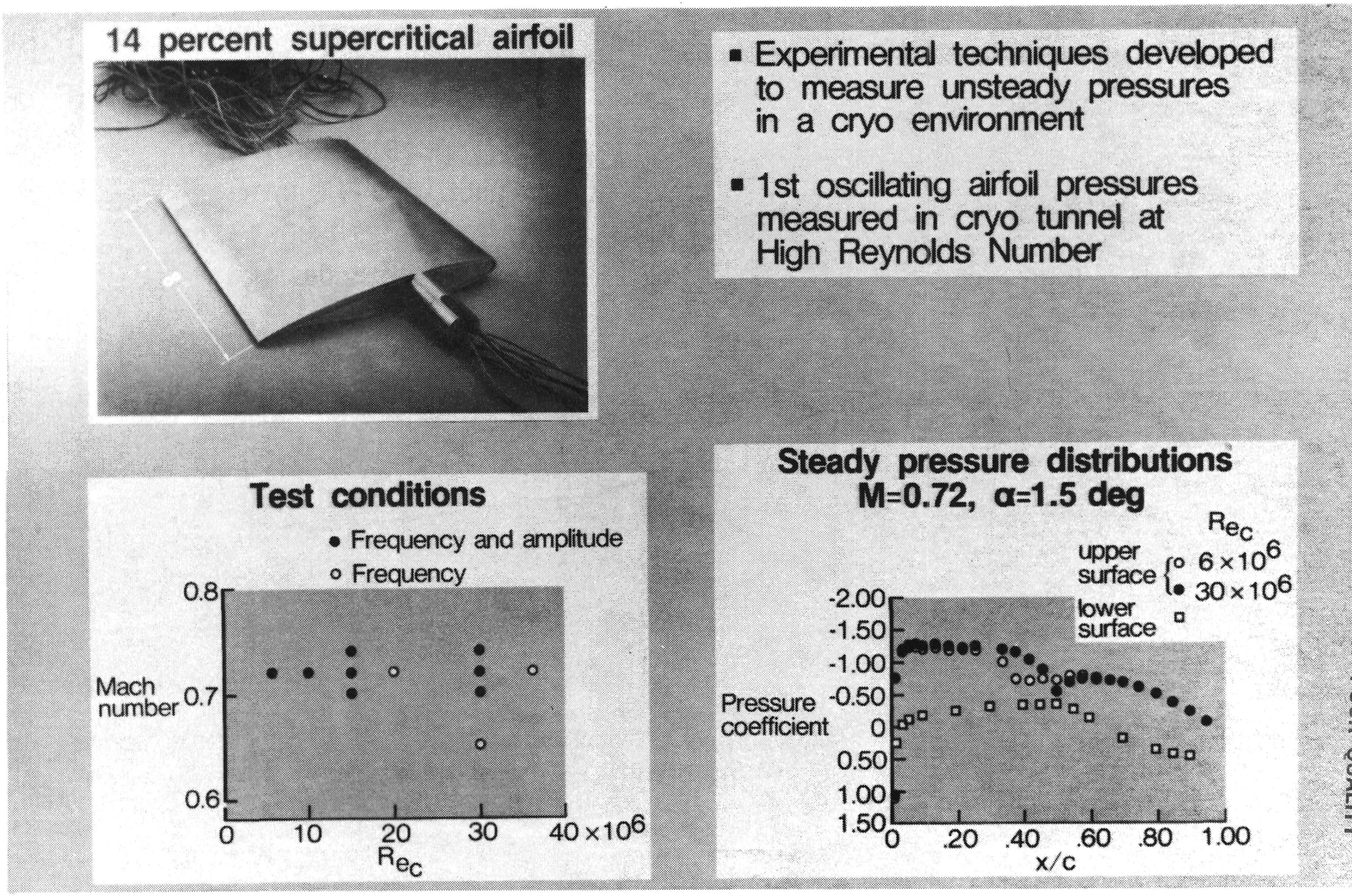


Figure 24. Reynolds number effects on unsteady pressure studied in the 0.3-meter Transonic Cryogenic Tunnel.

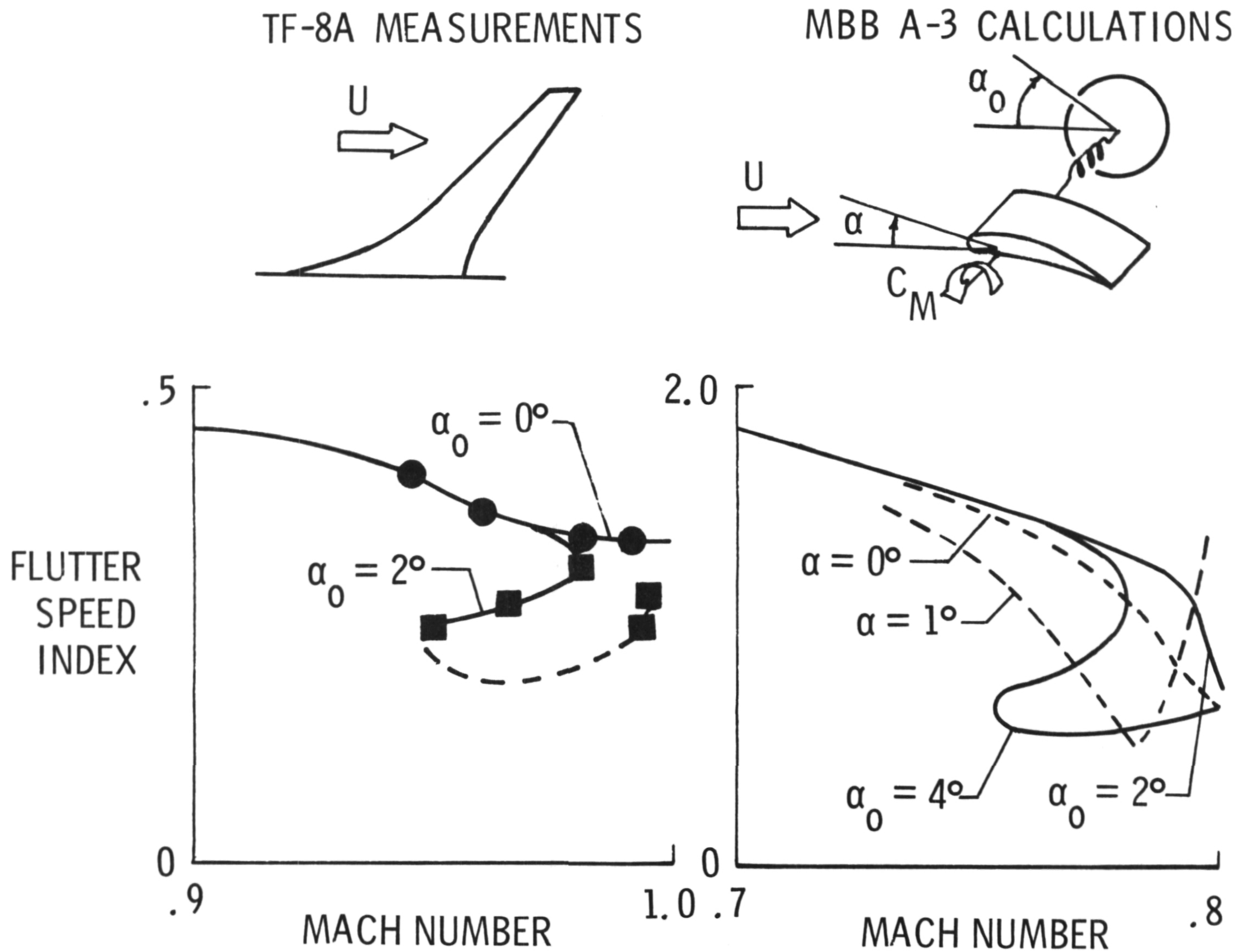


Figure 25. Transonic 2-D flutter calculations exhibit measured angle-of-attack effects.

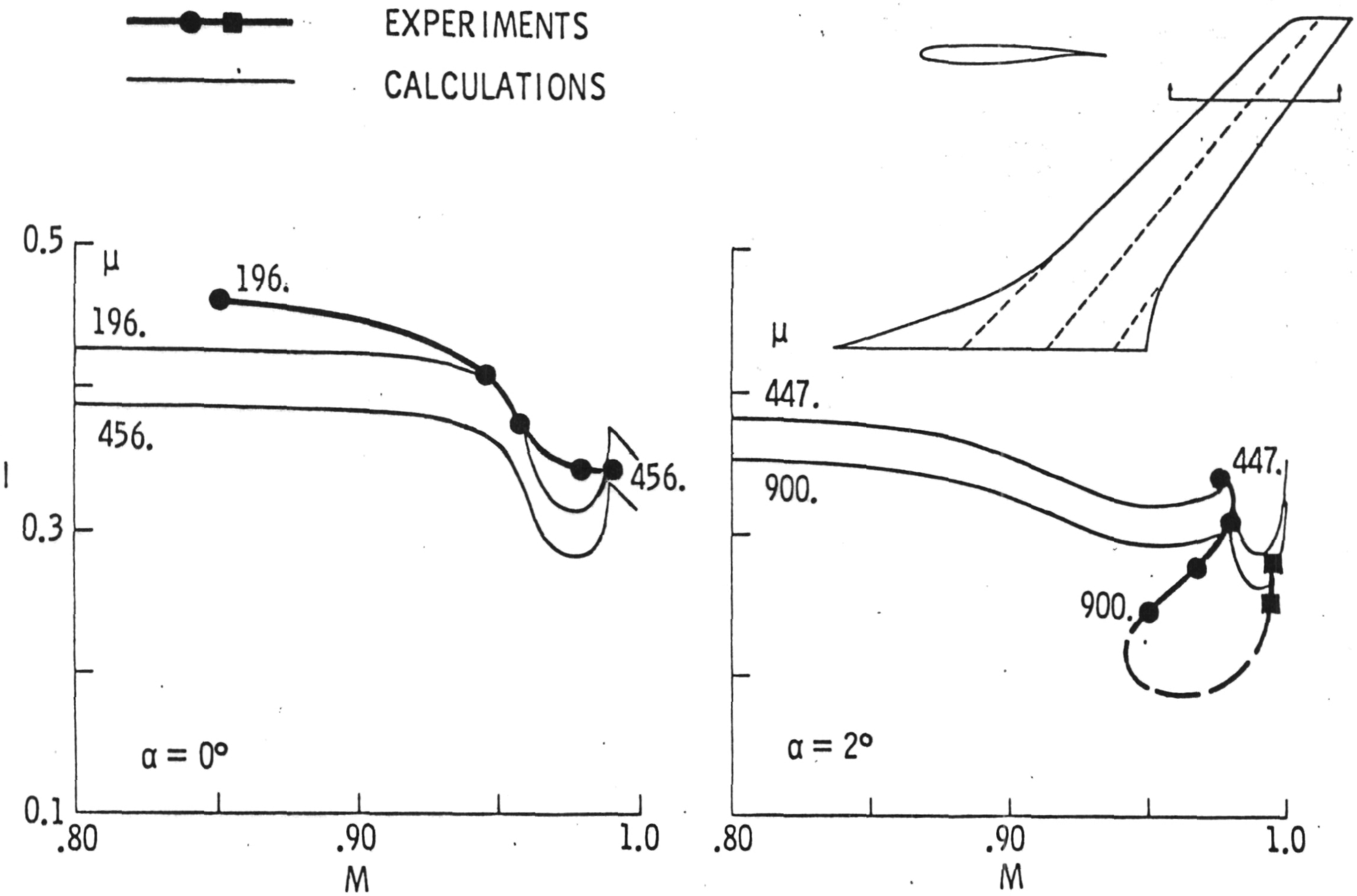


Figure 26. Experimental angle-of-attack sensitive flutter studied with modified strip analysis.

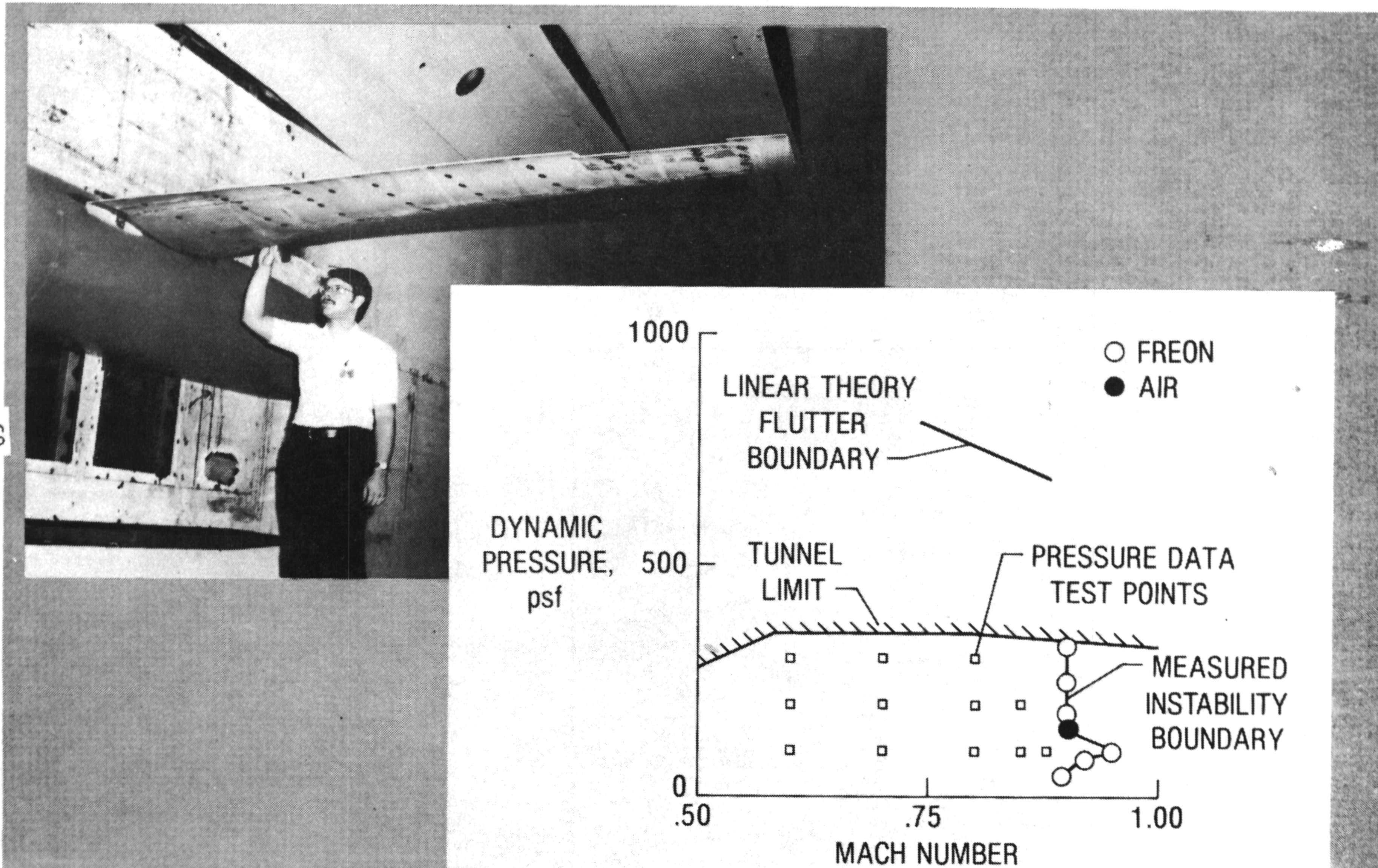


Figure 27. Pressure measurement tunnel conditions and unexpected instability boundary of a flexible wing study in the TDT.

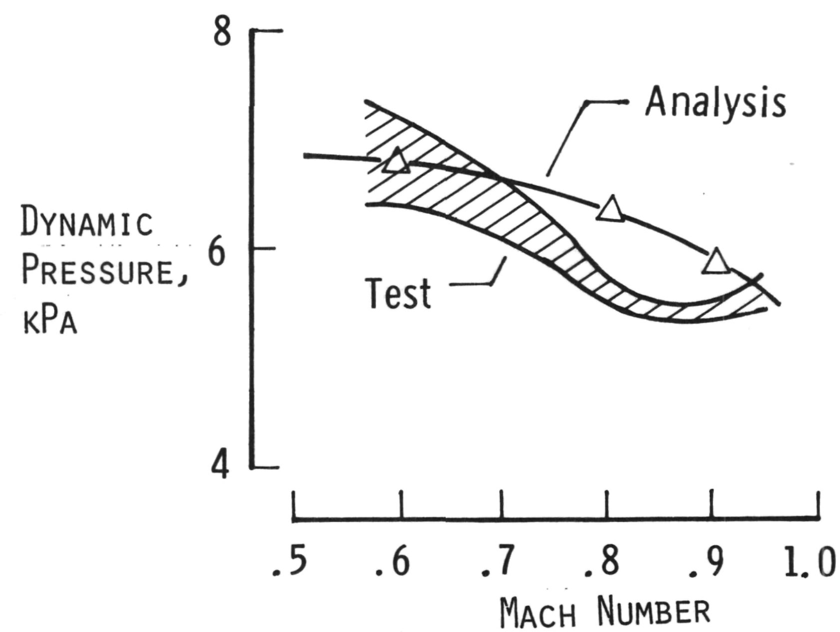


Figure 28. Calculated and experimentally predicted body-freedom flutter boundaries of a forward swept wing model.

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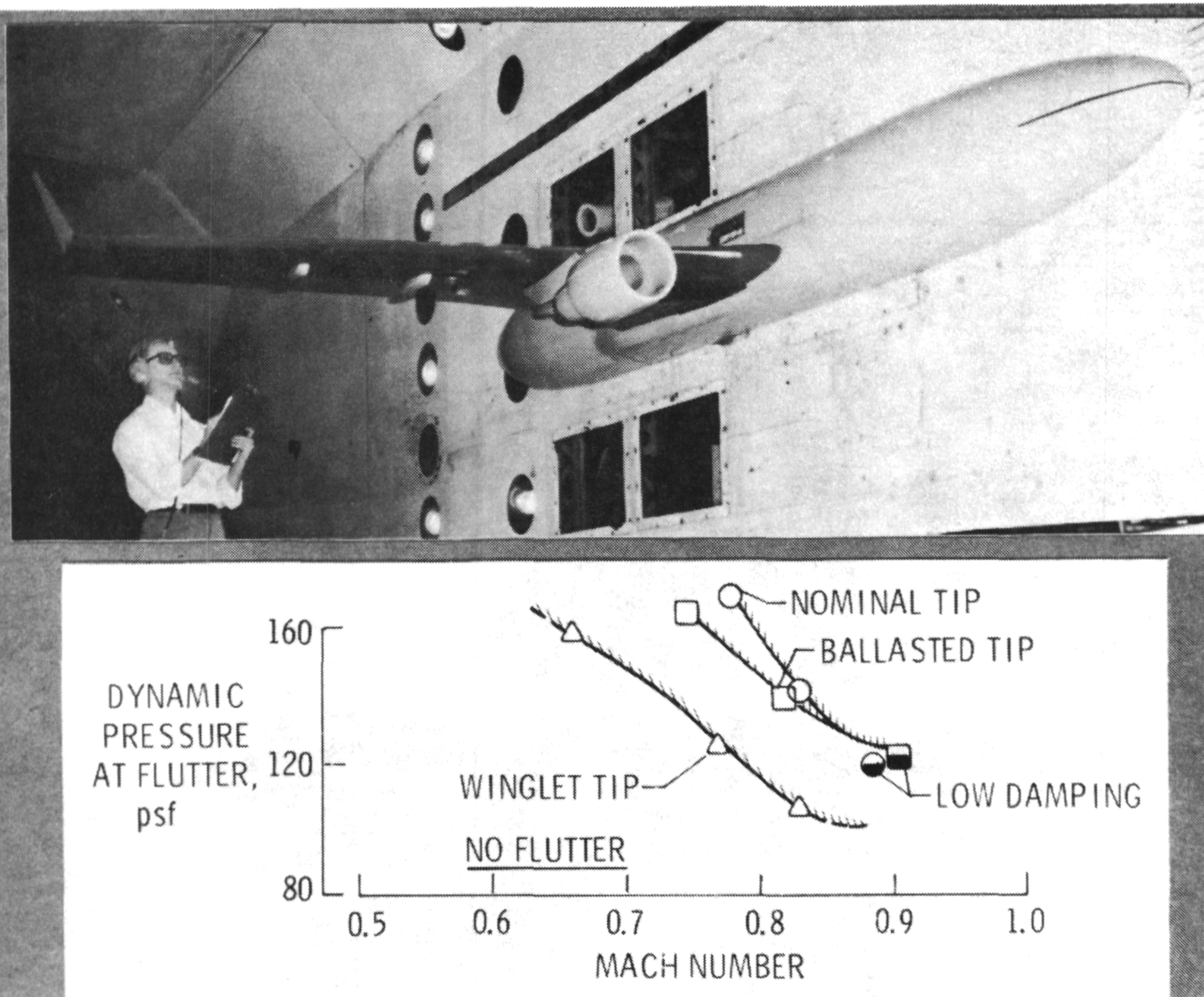

Figure 29. Effects of winglets on transonic flutter characteristics of a twin-engine transport type wing.

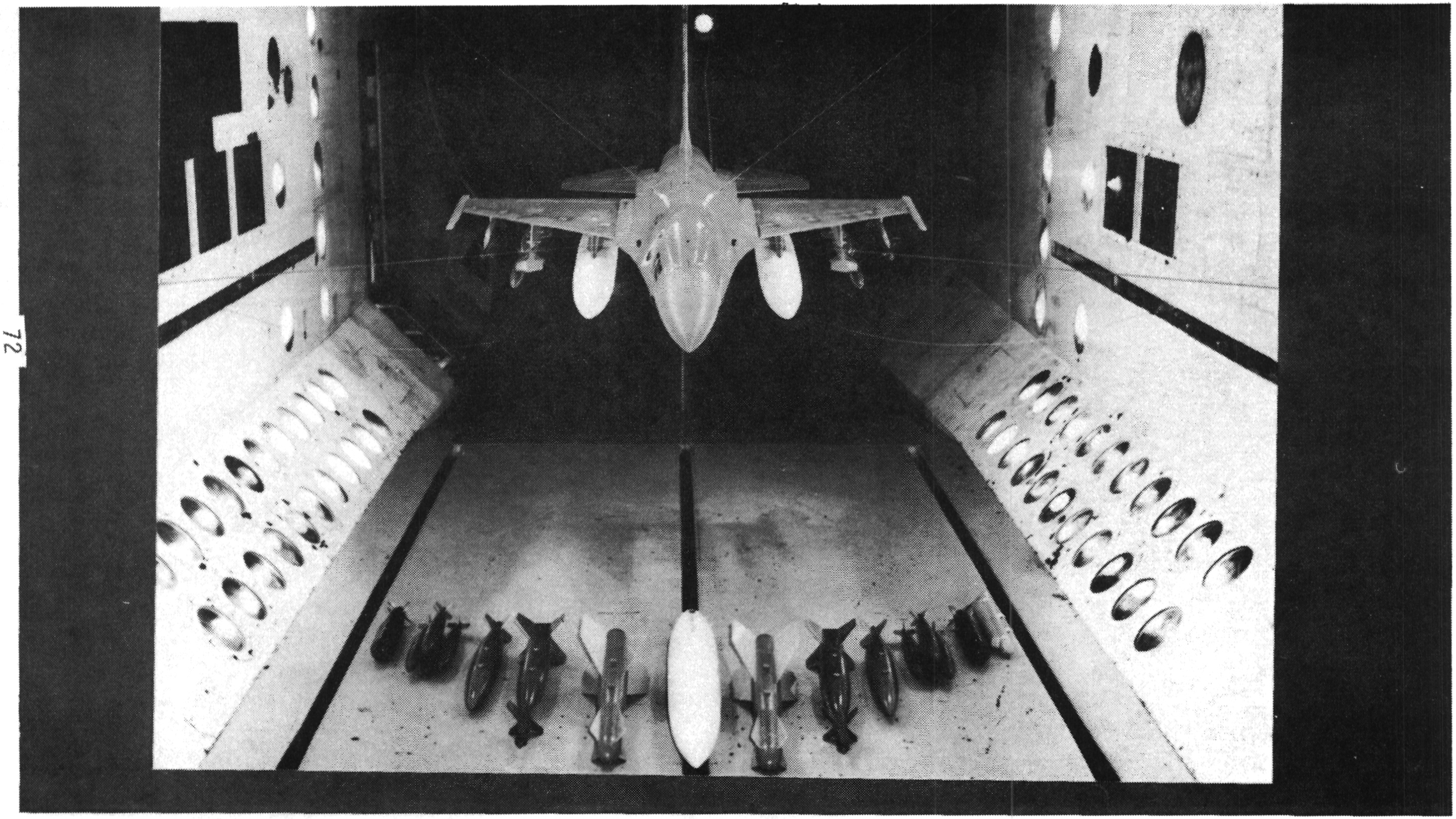

Figure 30. Effects of new fuel tanks and non-jettisonable pylons on F-16 flutter characteristics determined in TDT.

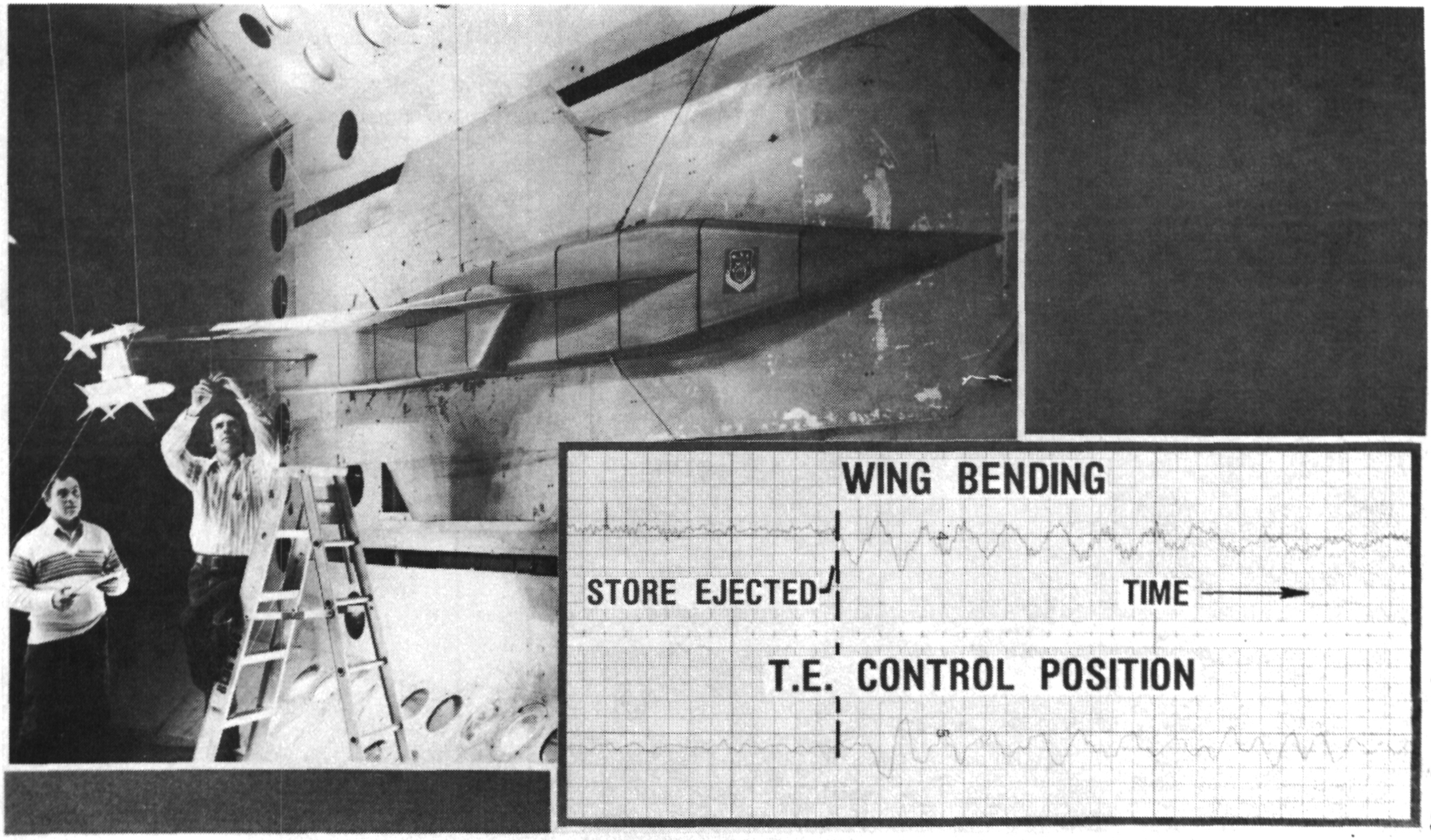

Figure 31. An adaptive digital active flutter suppression system is demonstrated in TDT tests.

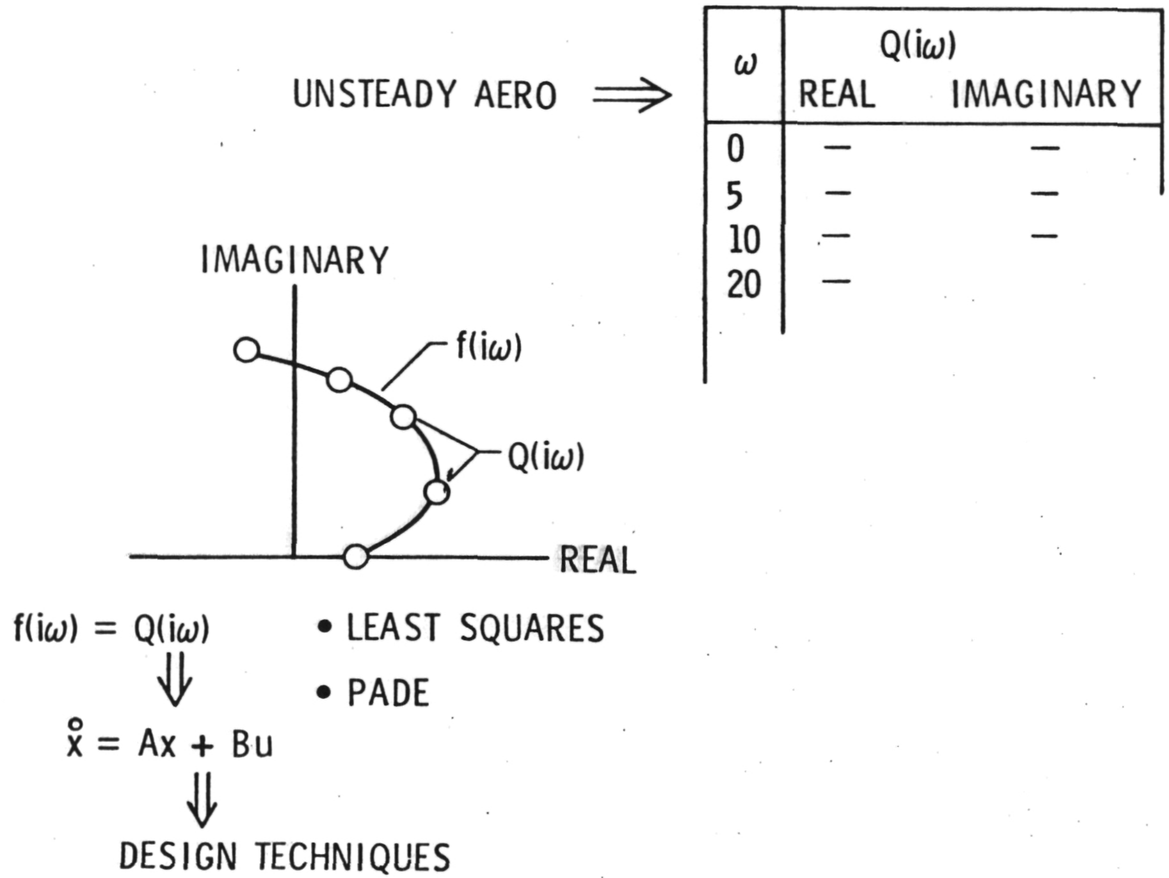

Figure 32. Method for incorporating unsteady aerodynamics in multidiscipline computer programs for analysis and design of active controls.

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16. Abstract

Recent progress in aeroelasticity, particularly at the NASA Langley Research Center is reviewed to look at the questions answered and questions raised, and to attempt to define appropriate research emphasis needed in the near future and beyond. The paper is focused primarily on the NASA Langley Research Center (LaRC) Program because Langley is the "lead" NASA center for aerospace structures research, and essentially is the only one working in depth in the area of aeroelasticity. Historical trends in aeroelasticity are reviewed broadly in terms of technology and staffing particularly at the LaRC. Then, selected studies of the Loads and Aeroelasticity Division at LaRC and others over the past three years are presented with attention paid to unresolved questions. Finally, based on the results of these studies and on perceptions of design trends and aircraft operational requirements, future research needs in aeroelasticity are discussed.

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