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LANGLEY AIRFOIL-RESEARCH PROGRAM

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INTRODUCTION

The purpose of this paper is to give an overview of past, present, and future airfoil research activities at the Langley Research Center. The immediate past and future occupy most of the discussion; however, past accomplishments and milestones going back to the early NACA years are dealt with in a broad-brush way to give a better perspective of current developments and programs. Indeed the seeds of the current surge in activity were sown a dozen years ago; the pedigree of many of Langley's present-day facilities can be traced to the mid thirties. In addition to the historical perspective, a short description of the facilities which are now being used in the airfoil program is given. This is followed by a discussion of new airfoil developments, advances in airfoil design and analysis tools (mostly those that have taken place over the past 5 or 6 years), and tunnel-wall-interference predictive methods and measurements. The last subject to be treated is future research requirements.

HISTORICAL PERSPECTIVE

Airfoil research at the Langley Memorial Aeronautical Laboratory began shortly after it was established and long before its first tunnel became operational in 1920. As assessment of the state-of-the-art of airfoil technology in the world was made and the airfoil data collected were put in a unified format and published for the benefit of the scientific community in the 1920, 1921, and 1923 NACA annuals (refs. 1 to 3). A few of these early airfoils are shown in figure 1; they indicate in a graphic way the lack of understanding of flow physics that existed in those early days. The airfoil sketched at the top is the USA 1 tested in an MIT tunnel at 13 m/sec (44 ft/sec) and is very similar to the Spad, Sopwith, Italian 2, and Eiffel 53 airfoils. The next two were tested at the Eiffel Laboratory in 1914 and were apparently designed to determine whether the performance of airfoils which are in fact two or three airfoils connected together would be superior to single-hump airfoils. Eiffel 44, the fourth from the top, has what appears to be a separation step on the top side; the philosophy behind its design is somewhat more obscure. It should be noted that the Eiffel Laboratory was probably the leading airfoil research center in the world prior to World War I and dozens of excellent airfoils were produced. The first Langley airfoil, which appeared in the 1923 NACA annual, was the Langley Memorial Aeronautical Laboratory 54. It was tested in the first NACA wind tunnel, the WT-1 5-foot tunnel, which, as noted earlier, began operating in 1920.

Reynolds number scaling was already a serious concern in 1920 and was the motivating factor in the design and construction of the variable density

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tunnel (VDT). When it came on-line in 1923. it was the most advanced tunnel in the world. It could operate at pressures up to 20 atmospheres and achieve Reynolds numbers up to 3.3×10^6 for a 13-cm (5-in.) chord model. Another facility of note that was constructed in the late twenties was the 12-inch high-speed tunnel which could produce velocities up to 350 m/sec (800 mph).

The latter half of the twenties saw the development of the NACA 4-digit and modified 4-digit series of airfoils. A sketch of one of the most popular, the 4412, is given in figure 2. A calculative procedure for airfoils of arbitrary shape was also formulated and provided a more rational basis for airfoil-family design.

Several more airfoil families were generated in the thirties including the 5-digit series, the 1-series (also known as the 16-series), and the NACA laminar-flow airfoil family. A 5-digit and a laminar-flow airfoil section are depicted in figure 2 along with a GA(W)-2 for comparison. The laminar-flow series was preceded by a considerable amount of research on calculative methods for boundary layers and empirical transition criteria.

Wind-tunnel construction during the thirties was driven by a desire to test airfoils under conditions approaching the flight environment, especially those designed for extensive runs of laminar flow. The low turbulence tunnel (LTT) started running in the mid thirties and, in addition to providing useful data, served as a learning experience for the design of the Langley low-turbulence pressure tunnel (LTPT). The LTPT started operation in 1941 and is still in use.

By the mid forties, the characteristics of a large number of airfoils had been defined; design procedures were being relied on for definition of airfoil geometry and modifications to existing airfoils. A report summarizing most of these data and procedures was published in 1945 (ref. 4). This document later became the basis for a book by Ira H. Abbott and Albert E. von Doenhoff (ref. 5). The late forties saw some shift in emphasis away from airfoil-section development to high-lift systems and boundary-layer transition and control.

A change in priorities in the early fifties necessitated a reduction in airfoil research, and by the mid fifties, it had completely disappeared. Airfoil research did not commence again until 1965 when R. T. Whitcomb conceived a new type of airfoil section for high subsonic speed applications (ref. 6). By virtue of a unique top side shaping, these airfoils were able to delay the formation of shocks and hence, for a given thickness, increase the drag-rise Mach number. Since these airfoils are able to sustain large regions of supercritical flow without shock formation near their design condition, they have been termed "supercritical" or "shockless" airfoils.

The first Whitcomb supercritical airfoil was a two-element design; it was followed closely by a single-element concept. Its value was quickly grasped by the aircraft industry and demands for supercritical airfoils for a variety of flight conditions and applications soon followed. This demand and interest led to a number of actions. It was decided in 1968 to reactivate Langley's airfoil test facilities; a series of supercritical airfoils were designed; theoretical efforts to provide a design method were commenced; and flight demonstration projects using the T2-C and F8V-1 aircraft were instituted. The first dividends from these actions came in the 1970-71 time period when flight tests clearly validated the airfoil concept and the Langley 6- by 19-inch transonic tunnel became operational. The first design and analysis codes for supercritical airfoils were published in 1972. These codes, developed at the NYU Courant Institute by a team of scientists led by P. R. Garabedian (ref. 7), are now in use all over the world.

The breakthrough in the design of airfoils for high subsonic applications precipitated a renewed effort to develop improved airfoils for low speeds. With the aid of a computer code developed at Lockheed under Langley contract by J. A. Braden, S. H. Goradia, and W. A. Stevens (ref. 8), R. T. Whitcomb designed the GA(W)-1 airfoil. It was tested in 1972 and showed a potential for better climb characteristics than airfoils in common use. Requests for the design of, and data on, similar airfoils with different thickness ratios were numerous.

By 1973, the number of Langley researchers and organizations engaged in airfoil research to explore the new concepts had increased in response to industry pressures for more data. Informal communication and individual planning no longer provided the coordination required of such a large effort. Consequently, in 1973, airfoil-research activities were programmized. Since that time, progress in the formulation of new airfoil designs and computer codes with improved capabilities has been remarkable. Some of the major milestones of the past 4 years are as follows:

Langley 6- by 28-inch transonic tunnel and 0.3-meter transonic cryogenic tunnel became operational.

Low-speed, medium-speed, and new supercritical airfoil families were defined.

Low-drag general aviation airfoils were developed.

New airfoil design and analysis methods were formulated.

Theoretical and experimental research on massive separation began.

Theoretical and experimental studies on wall interference were initiated.

General aviation airfoil design and analysis service was created.

With the exception of the airfoil design service, these items are discussed in the subsequent sections. The airfoil design service, created in 1976 at Ohio State University, gives the general aviation industry access to the latest computer codes for airfoil design and analysis. Scientists, expert in the application of all NASA-developed codes, are available to render whatever level of service is required.

TEST FACILITIES

The NACA and its successor, the NASA, have always been in the forefront of wind-tunnel technology and test techniques. Tunnel facilities developed over the past few years for airfoil tests and those being used in the program that were constructed many years ago continue to provide a unique capability for two-dimensional testing. The four wind tunnels which are used primarily for airfoil research are:

Langley low-turbulence pressure tunnel
Langley 6- by 19-inch transonic tunnel
Langley 6- by 28-inch transonic tunnel
Langley 0.3-meter transonic cryogenic tunnel (20- by 60-cm test section)

The Langley 8-foot transonic pressure tunnel has also been used for airfoil tests but this was done prior to the 6- by 28-inch transonic tunnel and the 0.3-meter transonic cryogenic tunnel becoming available.

The low-turbulence pressure tunnel (LTPT), as noted earlier, began operation in 1941 and is still in regular use (see fig. 3). It is still superior to most tunnels in the world over its operating range, which is Mach numbers from 0.1 to 0.4 and Reynolds numbers (based on a 0.6-m (2-ft) chord) from 1.0×10^6 to 30×10^6. New general aviation airfoils are developed with the aid of this facility; low-speed characteristics of supercritical airfoils are also explored. In addition, the excellent flow quality of the LTPT makes it ideal to carry out research on airfoils designed for natural or controlled laminar flow. Planned improvements for this facility will enable it to obtain accurate data for very high-lift systems and for airfoils at higher angles of attack than is now possible.

The 6- by 19-inch transonic tunnel came on-line in 1971 and has been utilized for both routine airfoil tests and technique-development research. A cross-section drawing of this facility is given in figure 4. It is a blowdown-type tunnel with no independent control of Mach number and Reynolds number. A 15-cm (6-in.) chord model can be tested to Reynolds numbers of 4.5×10^6 up to a Mach number of 1.0. Future utilization of the 6- by 19-inch tunnel will be primarily in the area of technique development with special emphasis on wind-tunnel wall interference.

The workhorse facility of the airfoil research is the 6- by 28-inch transonic tunnel, depicted in figure 5. Its normal Mach number range is from 0.3 to 1.0 with Reynolds numbers up to 13.5×10^6 for Mach numbers above 0.5. Independent control of Mach number and Reynolds number is possible. The 6- by 28-inch tunnel is used for research on every type of airfoil, including high-speed general aviation, supercritical, propeller, and rotorcraft. In the near future, the capabilities of this facility will be enhanced by the installation of a dynamic "rig" to carry out unsteady oscillatory and dynamic motion tests. Sidewall boundary-layer-control plates will also be provided to increase the maximum angle of attack at which useful data can be obtained.

The most versatile airfoil facility is the 0.3-meter transonic cryogenic tunnel (TCT) equipped with its 20- by 60-cm test section. A photograph of this facility is given in figure 6 and shows the unusual geometry of the tunnel with the test section at the top and the return leg at the bottom. The sketches given in figure 7 illustrate this fact more clearly. The bottom sketch shown is the original three-dimensional octagonal test section, the one in the middle is the two-dimensional insert with schlieren system in place, and the top sketch is a self-streamlining two-dimensional section soon to be constructed.

The use of nitrogen at cryogenic temperatures gives rise to Reynolds numbers up to a factor 5 larger than those of a conventional air tunnel. It also allows one to control Mach number, Reynolds number, and dynamic pressure independently, a capability more important perhaps in three-dimensional than in two-dimensional testing, since aeroelastic and Reynolds number effects can be isolated. Reynolds numbers up to 50×10^6 can be obtained for a 15-cm (6-in.) chord model. Production testing in the TCT should commence in the fall after completion of a number of minor improvements.

Further details on the capabilities and plans for Langley's two-dimensional research facilities can be found in the paper by E. J. Ray in these proceedings (ref. 9).

Research aircraft have been used as test facilities as well as wind tunnels. An F8V-1 fighter and a T2-C trainer aircraft were provided with new wings to determine the performance of supercritical sections in flight. In both applications, the supercritical wings proved superior to the original ones.

Similar proof tests of the new general aviation airfoil sections have been conducted. The GA(W)-1 was tested on the Advanced Technology Light Twin-Engine (ATLIT) aircraft, originally a Piper Seneca, and the GA(W)-2 on the Beech Sundowner. The latter is shown in flight near Columbus, Ohio, in figure 8.

NEW AIRFOIL DEVELOPMENTS

Langley's airfoil research program involves a variety of airfoil types including

- Low-speed general aviation*
- Low-speed natural laminar flow
- Medium speed
- Transport-type supercritical*
- Large cargo supercritical
- Laminar-flow control (LFC) supercritical
- Helicopter*
- Fighter
- Propeller

By far, the most effort has been expended on the three airfoil types indicated by the asterisks, and most of the subsequent discussion treats the accomplishments and plans for these types. New designs for the other types of airfoils (except LFC supercritical) have in some cases just been tested and, in others, are awaiting test or fabrication. The LFC airfoil is being developed under the NASA Energy Efficient Transport, Laminar Flow Control Project and is listed above only for completeness.

Comparisons of experiment with calculated results from the best low-speed and supercritical-airfoil predictive methods have generally shown that the theories are accurate. Theoretically determined pressure distributions and lift and moment coefficients correlate well with experiment; absolute-drag levels are sometimes in poor agreement. However, predictions of relative-drag level and drag-rise Mach number can usually be relied on. A result of these observations is that a philosophy has been adopted to test only a few representative samples of each type of airfoil to establish the validity of the theory. This philosophy has been applied in the case of the new Langley low-speed airfoil family. Figure 9 shows the range of thicknesses and lift coefficients of the new designs as well as their status. Five airfoils have already been tested, and four others have been designed for future testing. These test results coupled with available computer codes will enable other airfoils of this type to be designed with complete confidence. Additional information on NASA's low- and medium-speed airfoils can be obtained from the McGhee-Beasley paper in these proceedings (ref. 10).

The matrix of shapes which constitute the NASA supercritical airfoil family is shown in figure 10. Design lift coefficients vary from 0 to 1.0, and thickness ratios from 0.02 to 0.21. This family is based on improved design procedures developed by R. T. Whitcomb and were defined by using the transonic Bauer-Garabedian-Korn analysis code (ref. 7). Only two designs have been tested to date; four others are in the planning stages. Many other designs are available for test, as indicated by the solid dots, but it is likely that only about half will be so honored.

Rotor-airfoil research has the same objectives as those for conventional airfoils, that is, the evaluation of new airfoil-design methodology by wind-tunnel and flight tests and development of improved sections. Unfortunately, their achievement is considerably more difficult with current methods. Analysis tools which apply for two-dimensional steady flow must, in some rational way, be applied to the rapidly changing, three-dimensional environment of a rotor blade where relative velocities may change from high subsonic to low subsonic, or reverse direction in one revolution. Results contained in the proceeding papers by G. J. Bingham, K. W. Noonan, and H. E. Jones and C. E. Morris, Jr. would seem to indicate that progress is being made in this area (refs. 11 and 12).

Three new rotor airfoils designed for tests on the AH-1G helicopter as well as in Langley's two-dimensional wind tunnels are shown in figure 11. Each has been designed by a different method: the first using a transonic hodograph equation solution technique, the second using a linear-potential-equation method with compressibility corrections (parametric crestline), and the third using the transonic full-potential-equation method (supercritical technology) of F. Bauer, P. Garabedian, and D. Korn (ref. 7). All three have been tested in both the Langley 6- by 19-inch and 6- by 28-inch transonic tunnels as well as in flight.

The total rotor-airfoil-development effort can best be judged by the fact that a total of 29 airfoils have been tested, 21 in the 6- by 28-inch tunnel and 8 in the 6- by 19-inch tunnel. Some of the participating organizations involved are

6- by 19-inch transonic tunnel
NASA
U.S. Army R&T Laboratories (AVRADCOM)
Bell Helicopter Textron
National Aerospace Laboratory (NLR), Netherlands

6- by 28-inch transonic tunnel
Bell Helicopter Textron
Boeing Co.
Hughes Aircraft Co.
Sikorsky Aircraft
Wortmann

THEORETICAL DEVELOPMENTS

The ability to design airfoils and to analyze flows about them has grown by leaps and bounds over the past 10 years. Aided and abetted by a new generation of computers and improved solution techniques, designers can now quickly analyze supercritical airfoils with and without shocks and multielement airfoils, taking into account viscous effects. Significant progress has also been made in the treatment of airfoils with regions of separated flow. Many of the advancements cited have come out of the Langley airfoil-research program and many more are in store.

Specific areas where significant progress has been made and/or effort is being applied are

Design and analysis codes
Shock/boundary-layer interaction
Trailing-edge interaction
Leading-edge bubble interaction
Separated flow
Multielement airfoil analysis
Unsteady flow

A short discussion of each of these topics follows.

Perhaps the most significant development in airfoil theory in the past decade was the formulation of the hodograph design and "circle-plane" analysis codes for supercritical airfoils by the Garabedian-Korn-Bauer team at the NYU Courant Institute (ref. 7). The original and improved versions of these two programs are in use around the world and constitute one of the key technology advances being utilized by the aircraft industry in the design of the next generation of commercial transports. Figure 12 shows an airfoil which was designed using both the design and analysis codes. The top side geometry depicted was arrived at through repeated runs of the design code; the bottom side resulting from these same calculations was further modified by using the analysis code to make successive changes in the bottom contour until the desired pressure distribution was obtained. The pressure distribution depicted in figure 13 was obtained from the analysis code for a Mach number of 0.73 and a lift coefficient of 0.60.

A second set of programs capable of solving both the analysis and design problems was put together by L. Carlson of Texas A & M about 2 years ago (ref. 13). Carlson solves the full potential equations in the physical plane on a Cartesian coordinate system. His method has one advantage over the hodograph approach of Garabedian in that it can design airfoils for input pressure distributions with shocks. An example of this feature is given in figure 13. The design program was given the pressure distribution defined by the dashed line with a shock at approximately the 75-percent-chord station. The computer program produced the airfoil shown at the bottom and the slightly modified pressure distribution given by the solid line. Inserting the derived airfoil shape into the analysis program produced the circles and triangles which are in nearly perfect agreement with actual design input (the solid line).

One of the most vexing problems in airfoil analysis is the determination of drag, particularly at high subsonic speeds when imbedded shocks occur and under separated flow conditions. Most of the boundary-layer routines in use today for calculation of boundary-layer displacement thickness and skin-friction drag do not apply in regions where strong interactions occur with the inviscid flow. Three of these interaction regions are being studied in the Langley airfoil-research program; these are shock/boundary-layer, trailing-edge, and leading-edge bubble interaction.

The shock/boundary-layer interaction problem has been studied under NASA grant at the University of Michigan for the past 3 or 4 years. The investigation started with an idealized laminar-boundary-layer/oblique-shock case and proceeded through a series of steps to the normal-shock/turbulent-boundary-layer problem discussed in these proceedings in a paper by A. F. Messiter and T. C. Adamson (ref. 14). The next step will be to take this localized analysis and patch it into one of the full-potential airfoil-analysis programs such as the one developed by F. Bauer, P. Garabedian, and D. Korn described earlier (ref. 7).

The trailing-edge interaction may be even more important from a drag standpoint since the last 5 to 10 percent of the airfoil is responsible for most of the error in drag predictions. Empirical fixes currently employed in the boundary-layer routines near the trailing edge are generally reliable in terms of pressure-distribution predictions but are not consistent for drag estimates. R. E. Melnik of Grumman Aerospace Corporation has carried out a detailed analytical treatment of the trailing-edge interaction which holds promise of improved drag prediction. He has found that accounting for the effect of wake curvature is crucial, and an airfoil analysis computer code, due to A. Jameson, has been modified to include this effect. A pressure distribution made using this code is shown in figure 14. It is for the Korn 0.75 airfoil at a Mach number of 0.7 and a section lift coefficient of 0.669. Theoretical and experimental pressures are clearly in excellent agreement; the drags are not. The theoretical drag was 0.0082 as compared to an experimental value of 0.0107. There is some opinion that the experimental value is too high by about 20 counts, but this cannot be verified. (There is a general concern about most experimental drag data.) Drag correlations made using data on a GA(W)-2 airfoil at supercritical speeds obtained in an Ohio State University wind tunnel with a divided plenum are quite good. More research is required to obtain or identify "interference free" experimental data. Further details of the trail-- -edge-interaction methodology can be found in the paper by R. E. Me.. these proceedings (ref. 15).

Theoretical treatment of the leading-edge bubble interaction has been attempted by W. R. Briley and H. McDonald (ref. 16) using an iterative technique whereby the pressure is prescribed and boundary-layer profiles and displacement thicknesses are determined. The pressure is recalculated for the effective shape, and a new pressure input is formulated based on differences between the old and new pressures. In a low-lêvel effort, the same type of problem is being attempted at Langley using a different procedure whereby the displacement thickness is prescribed. This procedure, which has been developed by J. E. Carter and S. F. Wornom of Langley (ref. 17), is thought to have certain advantages over the pressure-prescribed method. It has been successful in calculating separated-flow bubbles in a depression and at the juncture of an afterbody-sting combination.

Airfoil flows with large amounts of separation have been calculated for a number of years using linear methods and empirical assumptions related to the point of separation and the separated region itself. More recently, these flow problems have been attacked using both numerical time-asymptotic methods for the Navier-Stokes equation and finite-difference relaxation methods for various forms of the nonlinear-potential-flow equation. The latter are used with a boundary-layer routine which is applied up to the point of separation.

R. W. Barnwell of Langley was the first to extend the ideas developed using linear potential equations to the finite difference approach (ref. 18). In Barnwell's calculation the separation point was not solved for; it was prescribed. An extension of his approach, whereby the separation point is determined in the calculation, has been undertaken by L. Carlson of Texas A & M under NASA grant (ref. 19). Some preliminary results have been obtained and an example is shown in figures 15 and 16. These figures show the pressure distribution at an angle of attack of 18° and lift vs. angle of attack for the GA(W)-2 airfoil at a free-stream Mach number of 0.15. Note in figure 15 that the flow separates at about 60 percent of the chord. Also, it should be recognized that at an angle of attack of 18°, the airfoil is beyond its maximum lift. Further exploration of this technique is underway.

So far the discussion of theoretical developments has been constrained to single-element airfoils. Progress has also been made in the analysis of multielement systems. Several analytical methods have been developed around the country over the past 6 years, and their capabilities and limitations are fairly well known. The features of the multielement program developed by Lockheed for Langley (ref. 8), and later modified by Boeing to make it more efficient (ref. 20), are listed in figure 17. This is the same program described earlier in the "Historical Perspective" section as having been used by Whitcomb to design the GA(W)-1 low-speed airfoil. As can be seen in figure 17, the program computes all the quantities of interest for as many as 7 elements. Correlations of theory with experiment show that this computer code yields results of good accuracy up to the point of flow separation. Further improvements are contemplated to improve the accuracy of the drag prediction, including an improved slot-flow analysis and a trailing-edge-interaction patch. The former would be attempted by applying operator splitting methods to the Navier-Stokes equations.

A companion study using the time-averaged Navier-Stokes equations is being carried out under NASA grant at Mississippi State University (ref. 21). It has reached the point where a two-element, compressible, turbulent flow code is now being debugged. A mapping procedure is used to transform the airfoil elements and the external flow field onto the interior of a rectangle where the equations are solved. The coordinate system in the physical plane is shown in figure 18. Note that the coordinate grid seems to compress in regions where the most resolution is required. It is expected that this computer program when fully developed will provide an excellent bench mark by which more approximate and faster techniques can be judged, including those treating separated flows.

The ability to analyze two-dimensional unsteady transonic flows is very much inferior to what one can do with steady flows. This is natural since the unsteady nonlinear potential equation is much more difficult to solve. A number of procedures have been tried; the two developed under the Langley program which have had the most success are the nonlinear, small-disturbance solution of Weatherill, Ehlers, and Sebastian (Boeing) (ref. 22) and the full-potential-equation solution of Isogai at Langley. An example calculation from the Isogai code is given in figure 19. It is for a steady, high Mach number flow where data are available. Theoretical results from a purely inviscid calculation and one in which the boundary-layer displacement effect is included are plotted. The lack of a proper accounting of the shock/boundary-layer interaction is the probable cause for much of the disagreement.

An extension of the Isogai code to include the effect of an oscillating flap is discussed in a paper by R. M. Bennett and S. R. Bland in these proceedings (ref. 23).

TUNNEL-WALL INTERFERENCE

The discussion of wind-tunnel-wall interference research has been isolated in a separate section, apart from theoretical developments and facilities, because of its special character and importance. Langley research in this area involves both theoretical and experimental studies for the assessment and elimination of wall interference. A list of many of these activities follows:

Slotted walls
- Barnwell correction of slot parameter
- Parametric slotted wall study in 6- by 19-inch transonic tunnel
- Slot flow diagnostic surveys in 8-foot transonic pressure tunnel

Adaptive walls
- Flexible wall experiment in 6- by 19-inch transonic tunnel
- Flexible wall theoretical prediction
- Two-dimensional adaptive walls for 0.3-meter transonic cryogenic tunnel

Some of these are discussed in the following paragraphs.

R. W. Barnwell has done an exhaustive study of slotted-wall boundary conditions and has provided new insights into the deficiencies of older methods. With the aid of existing data, he has derived a semiempirical design method for slotted-wall tunnels. Some of the data utilized came out of a parametric experimental study conducted in the 6- by 19-inch transonic tunnel by Everhart and Barnwell and reported elsewhere in these proceedings (ref. 24).

Very little data are available on the details of the flow in, and adjacent to, tunnel wall slots. More is needed to enable a better assessment of viscous effects and homogeneous boundary-condition assumptions. An experimental investigation is being carried out in the 8-foot transonic pressure tunnel to provide some of the needed data.

Langley has had a cooperative effort in adaptive wall research with the University of Southampton, England, for several years (ref. 25). In-house the 6- by 19-inch transonic tunnel has been used to explore this technique. Calculations carried out by Newman (LaRC) and Anderson (DCW Industries) for comparison with the 6- by 19-inch tunnel tests are discussed in their paper included in these conference proceedings (ref. 26). All of these activities have contributed to the design of an adaptive-wall two-dimensional section for the 0.3-meter transonic cryogenic tunnel.

A third approach to the wall interference problem has been proposed in the proceedings paper by W. B. Kemp, Jr. (ref. 27). Through the use of pressures measured near the tunnel walls as boundary conditions in a tunnel flow analysis program, he is able to determine whether a flow is correctable and, if so, what the corrections should be. The method, in effect, eliminates the need for a detailed knowledge of slot flows, porous wall flows, and so forth.

FUTURE RESEARCH REQUIREMENTS

A recurring theme in much of the research discussed was the need to improve the accuracy of drag predictions. Existing codes must be modified or new codes created to include the effects of strong interactions and flow separation. Although not discussed previously, turbulence models are also a source of drag error and a strong effort is needed to improve them.

Good progress in the prediction of flows with large separation was indicated, but many of the techniques are new and require further exploration. In order to obtain accurate data to validate these theories, it is mandatory that sidewall treatments in two-dimensional facilities be implemented and refined.

There is a dearth of unsteady pressure data at supercritical speeds. In addition to classical oscillatory data, dynamic-stall and buffet-type flows must be simulated. Wall interference corrections for unsteady motions is an area that has hardly been scratched.

Improved wall correction procedures for steady flow are still a requirement; the use of measured wall pressures should be pushed. Streamline-wall test sections should be "hard wired" to computers so that the wall adjustments can be automated.

As knowledge of flow stability improves, more research will be carried out on natural laminar flow and laminar flow with suction. This will require, in many cases, improvements in tunnel flow quality and reduced tunnel noise. Continued improvement in analysis tools to account for new stability theories and data will also be necessary.

Full-scale Reynolds number data are always desired. Only a few facilities around the world can obtain the levels required for large-transport airfoil sections. Detailed comparisons of the data from these facilities are required to ferret out error sources; efforts to obtain boundary-layer diagnostic data should be increased.

Finally, it should be recognized that airfoils are used in a three-dimensional environment. Considerably more effort to include the effects of sweep and taper in the design of airfoil sections is needed. In addition, the different environments of the wing root, midspan region, and tip should be better defined so that airfoil sections can be designed taking into account these differences.

Clearly, there are many research opportunities and challenges in airfoil aerodynamics. If they are undertaken with the same enthusiasm as that applied during the past decade, then another quantum leap in airfoil-aerodynamics capabilities can be expected.

APPENDIX

SYMBOLS

In this appendix, symbols which are used on the figures are defined.

C_p	pressure coefficient, $\dfrac{p - p_\infty}{q_\infty}$
C_p^*	sonic pressure coefficient, $\dfrac{p^* - p_\infty}{q_\infty}$
c	airfoil chord
c_d	section drag coefficient, $\dfrac{\text{Drag force}}{q_\infty c}$
c_l	section lift coefficient, $\dfrac{\text{Lift}}{q_\infty c}$
c_m	section pitching-moment coefficient, $\dfrac{\text{Pitching moment}}{q_\infty c^2}$
M_∞	free-stream Mach number
p	local static pressure
p_∞	free-stream static pressure
p^*	static pressure at sonic velocity
q_∞	free-stream dynamic pressure
R	unit Reynolds number
R_c	Reynolds number based on airfoil chord
t	maximum airfoil thickness
V_∞	free-stream velocity
α	angle of attack
δ^*	boundary-layer displacement thickness
ρ_∞	free-stream density

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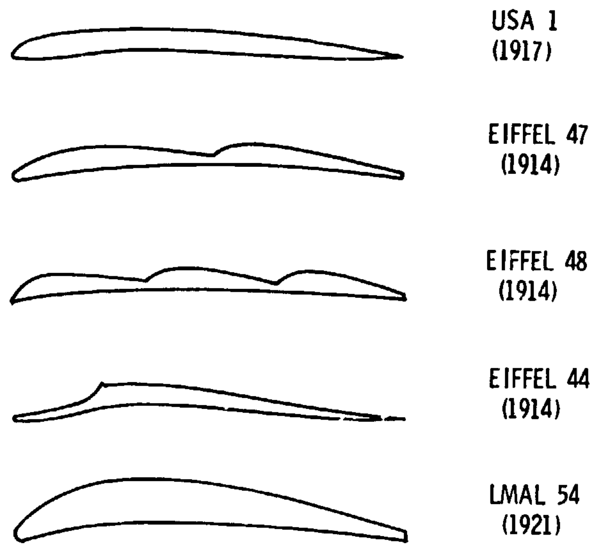


Figure 1.- Early airfoil shapes.

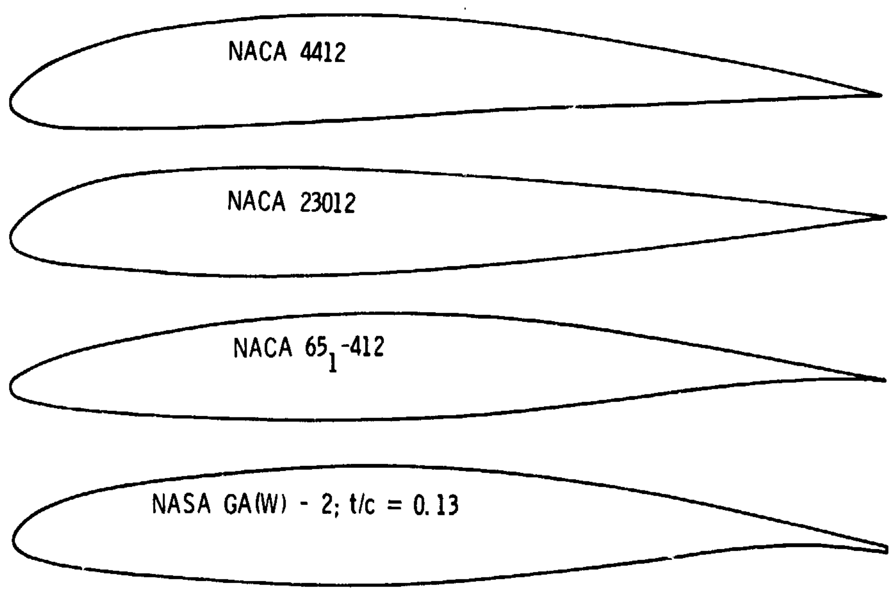


Figure 2.- Comparison of older NACA airfoils with the NASA GA(W)-2 airfoil.

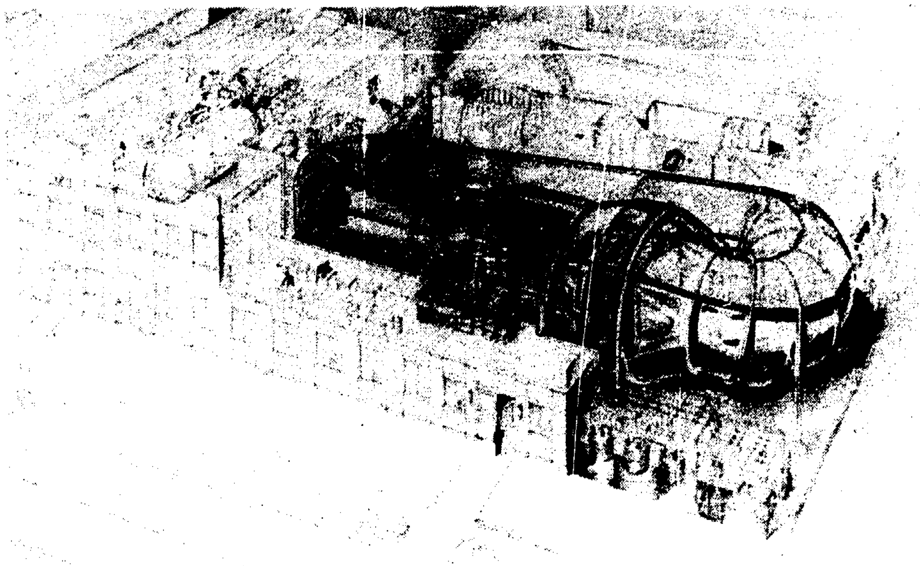


Figure 3.- The Langley low-turbulence pressure tunnel.

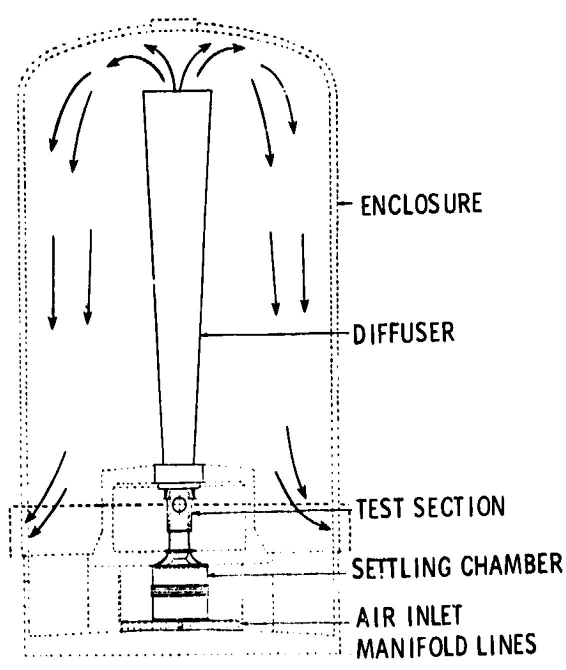


Figure 4.- Cross-section drawing of Langley
6- by 19-inch transonic tunnel.

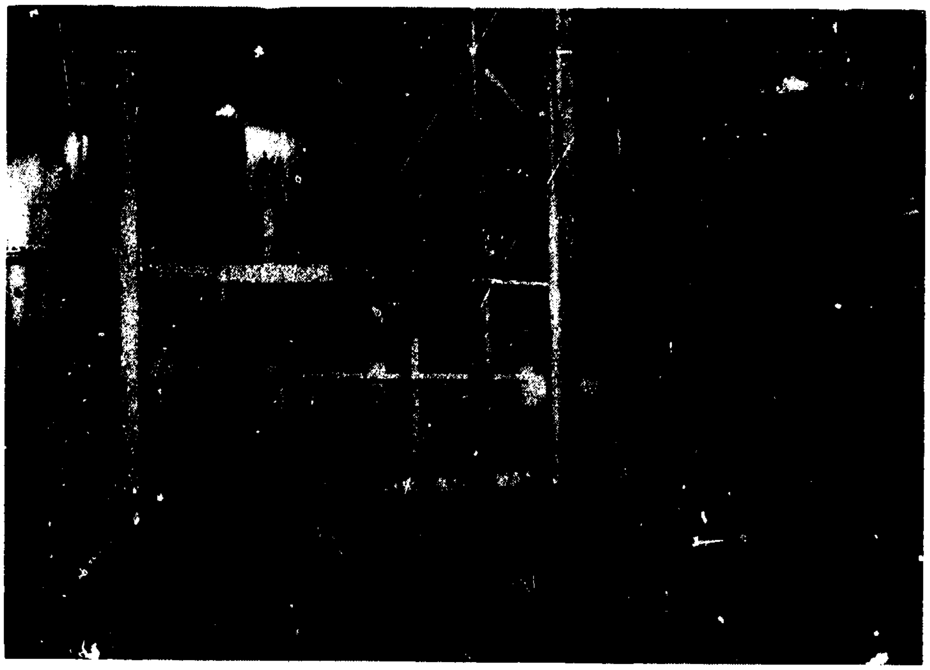


Figure 5.- Photograph of the Langley 6- by 28-inch transonic tunnel.

Figure 6.- Photograph of the Langley 0.3-meter transonic cryogenic tunnel with 20- by 60-cm two-dimensional test section.

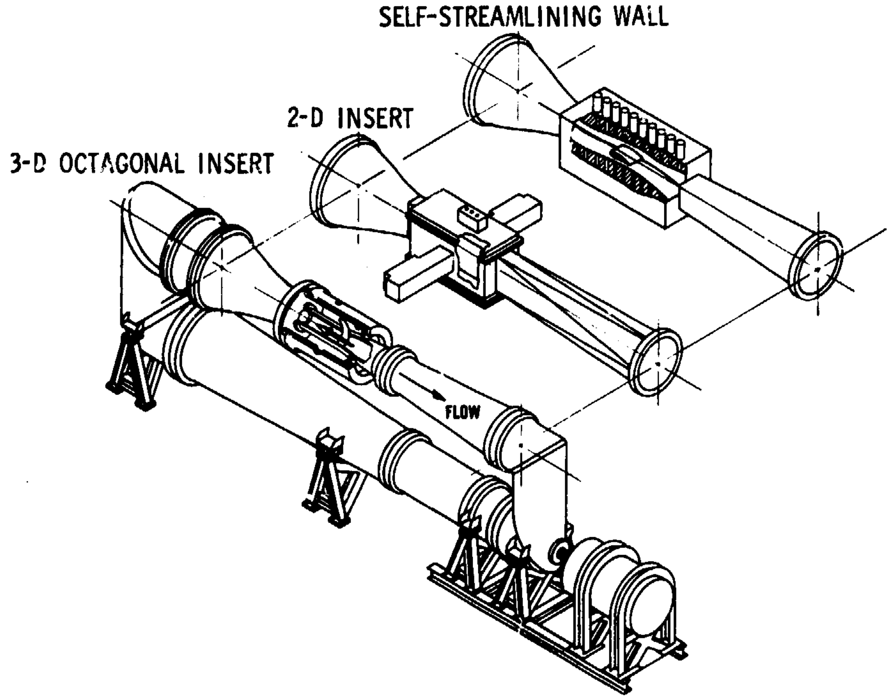


Figure 7.- Interchangeable test section capability of 0.3-meter transonic cryogenic tunnel.

Figure 8.- GA(W)-2 airfoil on Beech Sundowner.

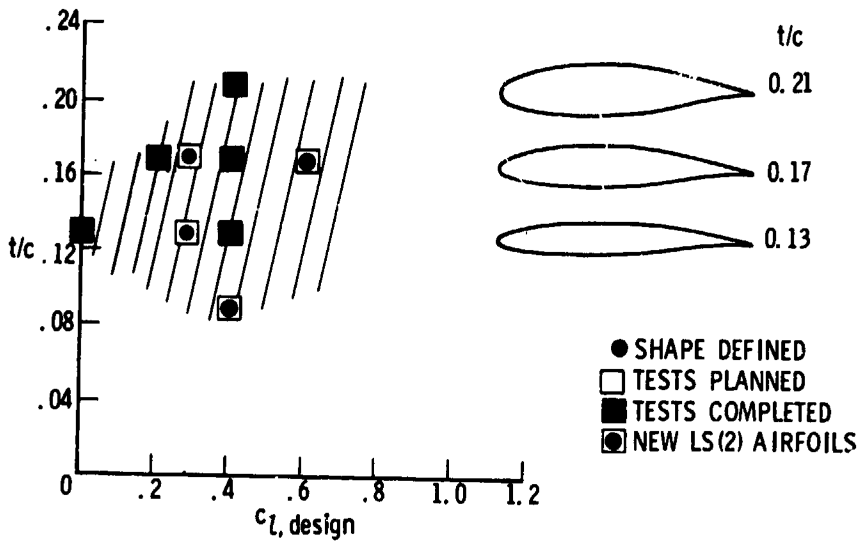


Figure 9.- Langley low-speed airfoil family.

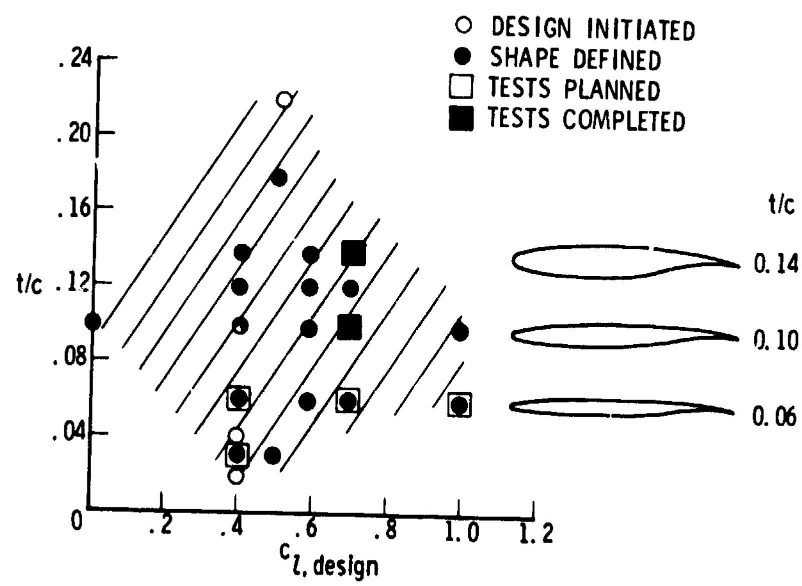


Figure 10.- NASA supercritical airfoil family.

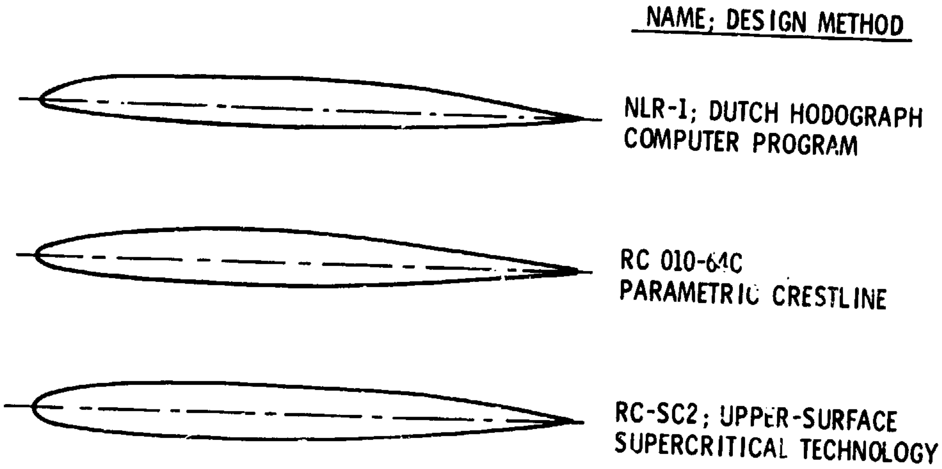


Figure 11.- Airfoils for AH-1G flight test.

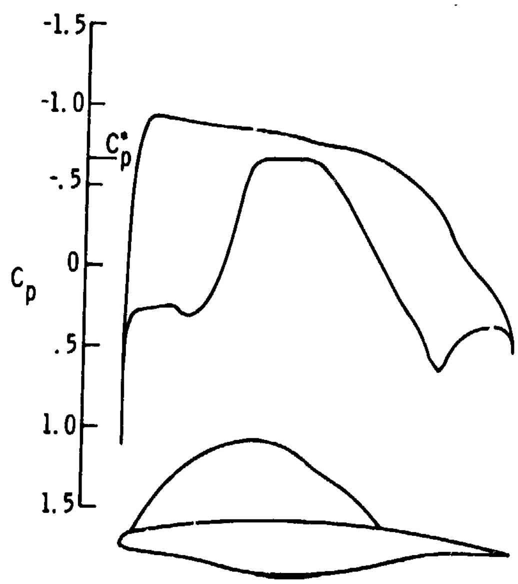


Figure 12.- Pressure distribution over an LFC airfoil calculated by the Korn-Carabedian analysis program. $M_\infty = 0.73$; $c_l = 0.60$.

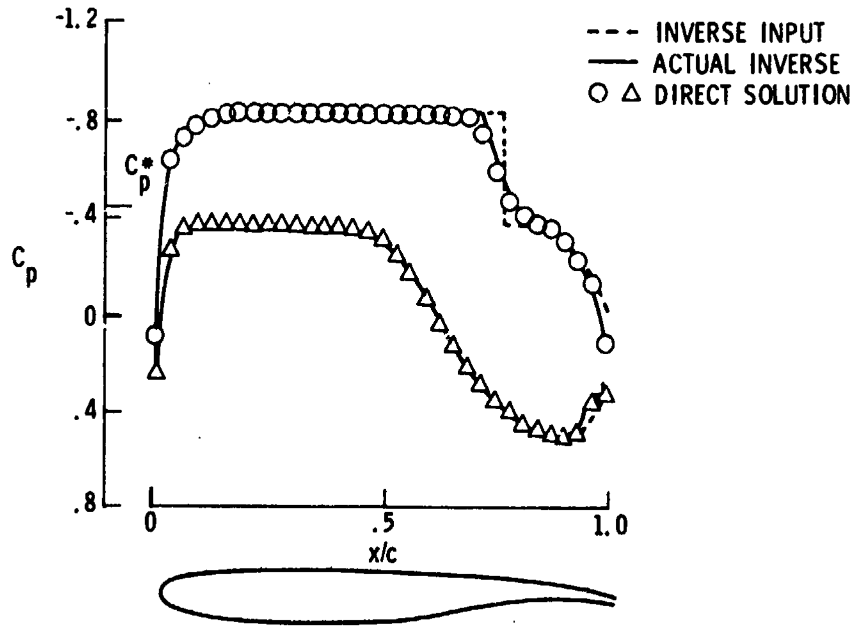


Figure 13.- Comparison of direct and inverse C_p distributions calculated by Carlson computer codes. $M_\infty = 0.80$; $\alpha = -0.5^\circ$.

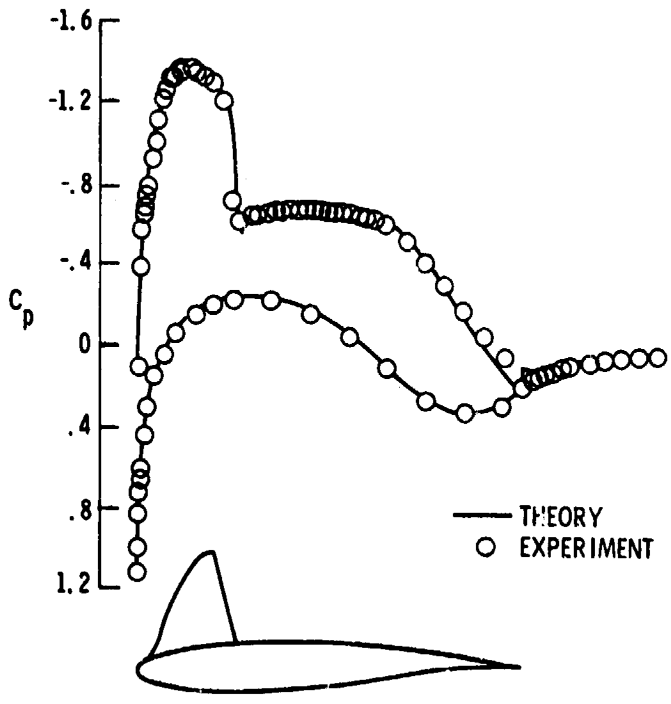


Figure 14.- Theoretical pressure distribution on Korn 0.75 airfoil including trailing-edge interaction. $M_\infty = 0.7$; $c_l = 0.669$.

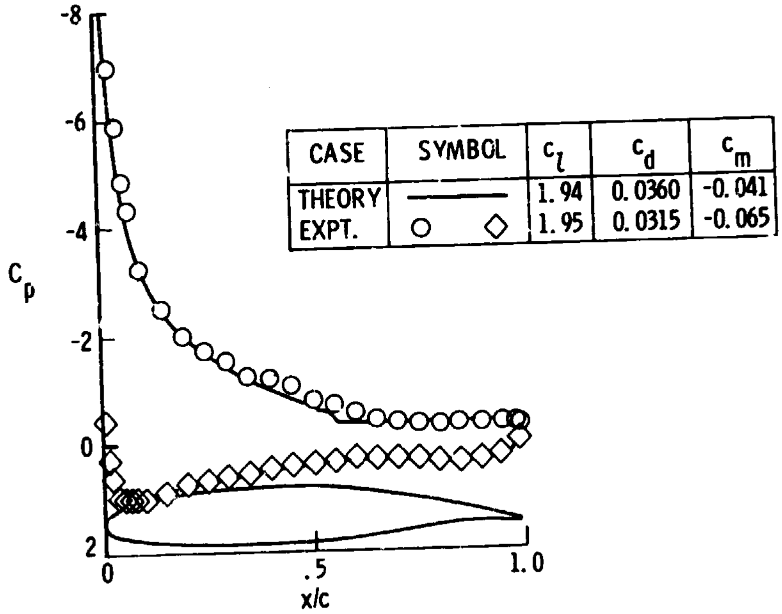


Figure 15.- Theoretical and experimental pressure distribution comparisons for GA(W)-2 airfoil with large separated flow region. $M_\infty = 0.15$; $\alpha = 18^o$.

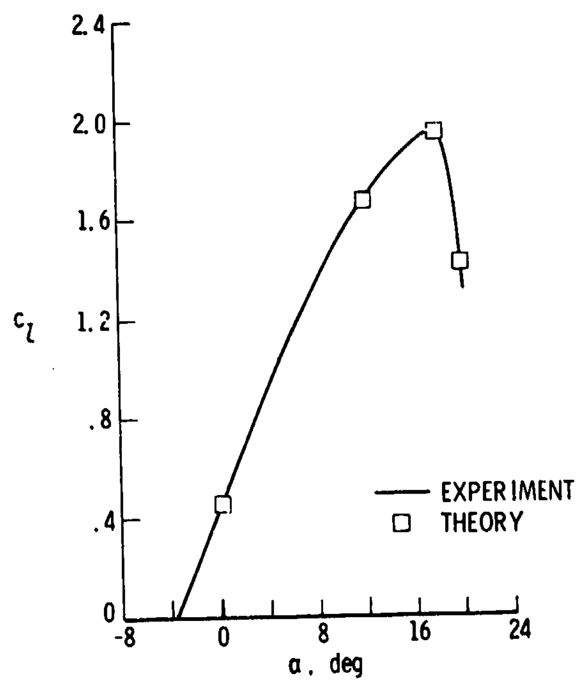
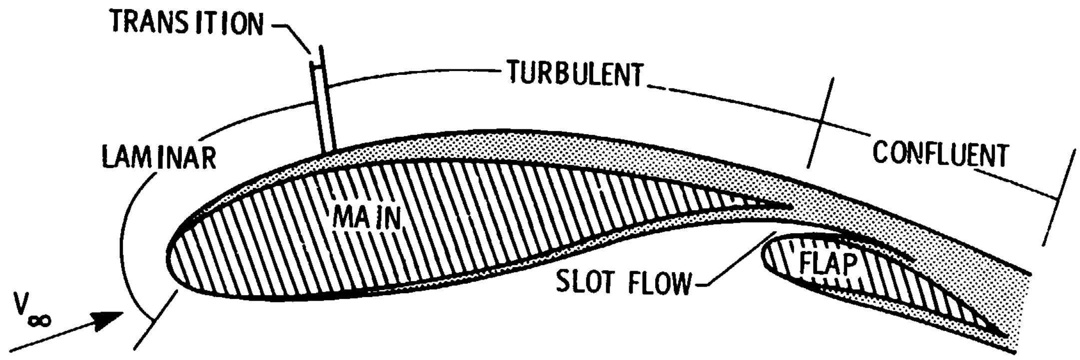


Figure 16.- Experimental and theoretical variations of c_l with α for GA(W)-2 airfoil.



CAPABILITIES

- 1 TO 7 COMPONENTS
- VARIABLE α, M_∞ AND R
- FIXED AND/OR FREE TRANSITION
- COMPUTES c_l, c_d, c_m, AND BOUNDARY LAYERS, C_p

PROGRAM FEATURES

- POTENTIAL-FLOW SOLUTION
- BASIC BOUNDARY LAYER
- SLOT-FLOW ANALYSIS
- CONFLUENT BOUNDARY LAYERS
- COMBINED SOLUTION

Figure 17.- Two-dimensional viscous-flow multielement airfoil program features.

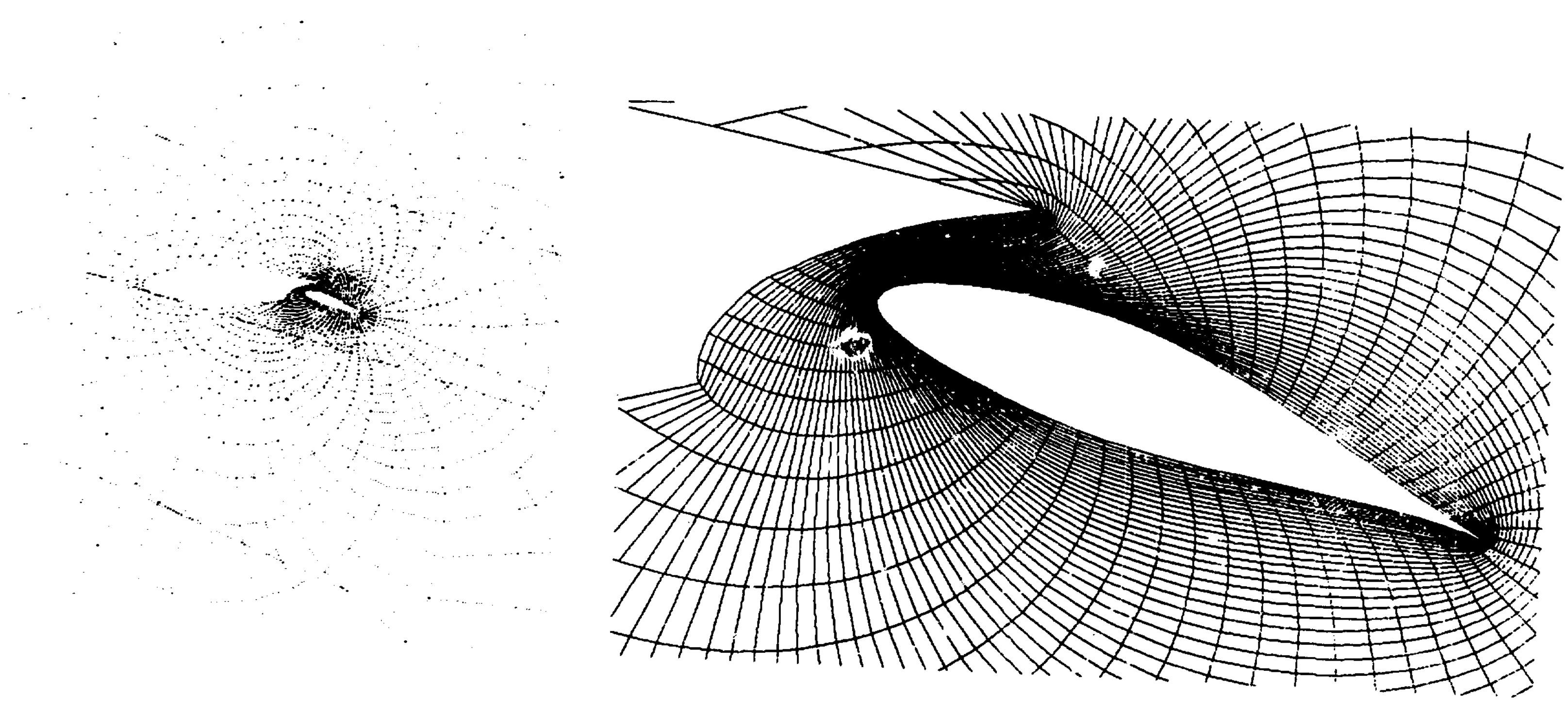


Figure 18.- Grid in physical plane for Navier-Stokes solution of multielement airfoils.

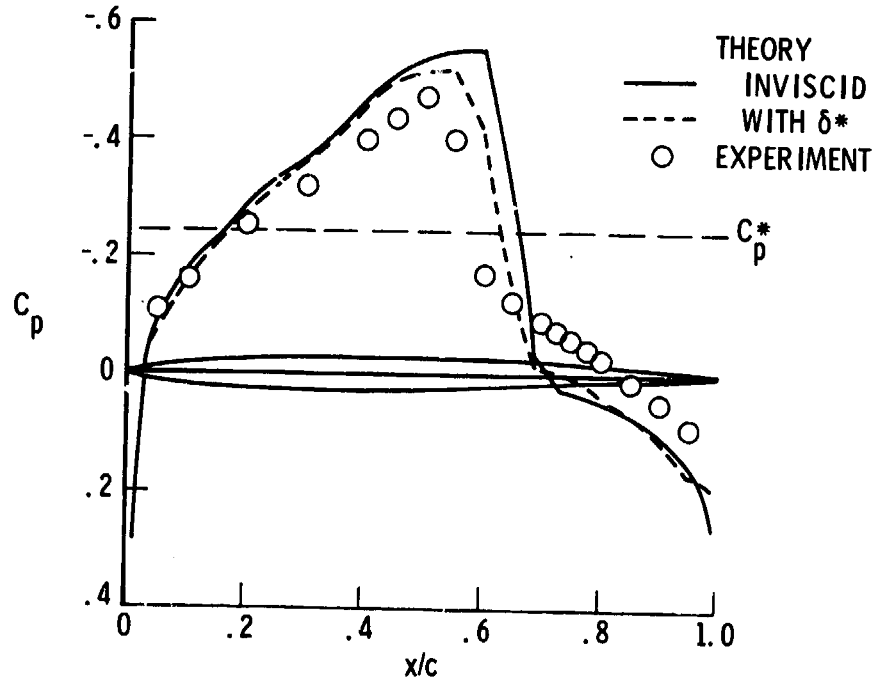


Figure 19.- Steady flow calculation for the NACA 64A006 airfoil using Isogai unsteady potential flow code. $M_\infty = 0.875$; $\alpha = 0$.