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COMPUTATIONAL DESIGN OF NATURAL LAMINAR FLOW WINGS

FOR TRANSONIC TRANSPORT APPLICATION

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ABSTRACT

Two research programs are described which directly relate to the application of natural laminar flow technology to transonic transport-type wing planforms. Each involved using state-of-the-art computational methods to design three-dimensional wing contours which generate significant runs of favorable pressure gradients. The first program supported the Variable Sweep Transition Flight Experiment and involves design of a full-span glove which extends from the leading edge to the spoiler hinge line on the upper surface of an F-14 outer wing panel. Boundary-layer and static-pressure data will be measured on this design during the supporting wind-tunnel and flight tests. These data will then be analyzed and used to infer the relationship between crossflow and Tollmien-Schlichting disturbances on laminar boundary-layer transition. A wing was designed computationally for a corporate transport aircraft in the second program. The resulting wing design generated favorable pressure gradients from the leading edge aft to the mid-chord on both upper and lower surfaces at the cruise design point. Detailed descriptions of the computational design approach are presented along with the various constraints imposed on each of the designs. Wing surface pressure distributions, which support the design objectives and were derived from transonic three-dimensional analyses codes, are also presented. Current status of each of the research programs is included in the summary.

NLF WING DESIGN PROGRAMS

There are two natural laminar flow (NLF) wing design programs in which the Applied Aerodynamics Group has been involved. The Variable Sweep Transition Flight Experiment (VSTFE) was formulated between NASA Ames-Dryden and NASA Langley to establish a data base on the effects of the interaction between crossflow (CF) and Tollmien-Schlichting (TS) instabilities on boundary-layer transition utilizing the F-14 aircraft as a test bed. This involved modifying the F-14 wing outer panel such that favorable pressure gradients could be generated over a wide range of flight conditions. Extensive computations were performed to verify the potential flow methods used in the effort and in the actual design of the wing outer panel.

The second program involved the design of an NLF wing for a proposed high-aspect-ratio, low-sweep, corporate-transport aircraft. Unlike the VSTFE, no data were available on the baseline configuration so the wing design was based totally on computational results. Another unique challenge posed by this design problem was that acceptable aerodynamic characteristics with a fully turbulent boundary layer over the entire wing had to be maintained.

- F-14 OUTER PANEL GLOVE IN SUPPORT OF THE VARIABLE SWEEP TRANSITION FLIGHT EXPERIMENT

- HIGH ASPECT RATIO, LOW SWEEP NLF WING

COMPUTATIONAL METHODS APPLIED TO NLF WING DESIGN

The computational methods that were applied to the NLF wing designs included two two-dimensional codes and three three-dimensional codes.

The New York University airfoil analysis code written by Bauer, Garabedian, and Korn, reference 1, is used extensively by researchers for two-dimensional flow analysis. The inviscid solution solves for steady, isentropic, irrotational flow. Viscous corrections are provided by adding the turbulent boundary layer to the airfoil contour. There is no laminar boundary-layer capability in the code.

The high-lift code, reference 2, is a subsonic panel code with an integral boundary layer. This code is used to evaluate the low-speed, high-lift characteristics of an airfoil.

TAWFIVE, reference 3, solves for the full-potential equation on wing-body configurations using conservative differencing. This code combines FLO-30 with a three-dimensional integral boundary layer. Solutions are obtained on a body-fitting grid which allows for an arbitrary fuselage to be modeled.

WBPPW, reference 4, solves for the flow field around wing, body, pods, pylons, and/or winglets. The code is characterized by a unique grid-embedding technique which provides excellent resolution around various configuration components. Using nonconservative finite-difference approximations, a modified small-disturbance equation is solved in the embedded grid system. Viscous corrections are provided through a two-dimensional strip boundary-layer method which adds displacement thickness slopes to the wing surface slopes.

The FLO-22NM code, reference 5, is the FLO-22 wing-alone code with the following improvements: a quasi-inverse design capability, corrections to the plane of symmetry boundary condition, incorporation of a potential-flow/boundary-layer interaction scheme, simulation of the wing-fuselage interference effect, and an improved drag computation methodology. The full-potential equation is approximated using nonconservative finite-difference techniques. Boundary-layer corrections are based on two-dimensional integral formulations. The quasi-inverse design mode of FLO-22NM is based on an approach by Garabedian and McFadden, reference 6. The wing contour is modified systematically to drive the wing surface pressure distribution toward a specified target pressure distribution.

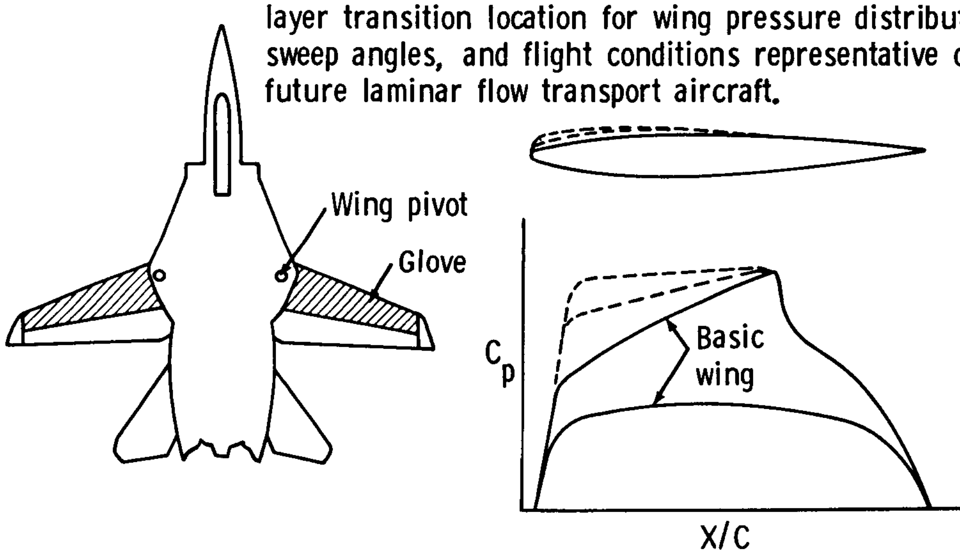
- TWO – DIMENSIONAL
  - NYU GARABEDIAN AND KORN
  - HIGH LIFT CODE
- THREE – DIMENSIONAL
  - TAWFIVE
  - WBPPW
  - FLO22NM

F-14 VARIABLE-SWEEP TRANSITION FLIGHT EXPERIMENT

An important question that must be answered in order to design wings which effectively utilize NLF relates to boundary-layer transition. It is known that, for boundary layers in a three-dimensional flow environment, there is an interaction between crossflow (CF) and Tollmein-Schlichting (TS) instabilities that can cause transition to occur in an otherwise favorable environment (i.e., favorable pressure gradient, smooth surface, etc.), reference 7. In order to assist in identifying and quantifying the influence of the CF-TS interaction on wing boundary-layer transition, data are needed for various combinations of favorable pressure gradients, Reynolds numbers, and sweep angles. This is the objective of the VSTFE. The F-14 aircraft was selected as the test bed aircraft because of its variable sweep capability, which would allow data to be taken over a wide range of sweep angles.

The approach of this flight experiment is to modify the wing outer panel by gloving on a foam and fiberglass contour so that favorable pressure gradients will be generated over a range of Mach numbers, sweep angles, and Reynolds numbers. Two different gloves were designed which correspond to a M = 0.70 and M = 0.80 design condition. NASA Langley was responsible for the M = 0.70 glove design, and Boeing Aircraft Company was responsible for the M = 0.80 glove design. Both gloves will be flown simultaneously, one on each wing of the F-14, resulting in an asymmetric configuration. Hence, a maximum constraint on the rolling moment because of the asymmetric configuration was imposed on the design.

Objective: Obtain accurate in-flight measurement of boundary layer transition location for wing pressure distributions, sweep angles, and flight conditions representative of future laminar flow transport aircraft.



INITIAL QUESTIONS

Extensive computations have been performed using small-disturbance and full-potential flow codes in the design of the NLF glove for the VSTFE. Several questions pertaining to computational modeling of the F-14, the applicability of two-dimensional codes to the design problem, and the ability of three-dimensional codes to accurately predict the flow field on the wing outer panel were addressed. In addition, it was necessary to evaluate the results on the subject configuration from the small-disturbance and full-potential codes.

The F-14 had two geometric characteristics which were the cause for concern with respect to modeling the configuration in the transonic analysis codes. The side-mounted inlet-fuselage combination and the highly-swept strake regions were thought to possibly have significant effects on the outer panel flow field. A sequence of runs was set up to answer these modeling questions using the WBPPW code which has the capability of modeling an arbitrary body. By comparing the results with experimental data the following observations were made (ref. 8):

> The highly-swept strake region has a significant influence on the wing outer panel pressures and therefore should be modeled.
>
> It is not critical to accurately model the inlets or nacelles.
>
> An axisymmetric body with an area distribution equal to the actual aircraft is adequate.

With these observations in hand, it was felt that an accurate configuration model could be developed for the full potential code, TAWFIVE, which would complement the analysis of the small-disturbance code.

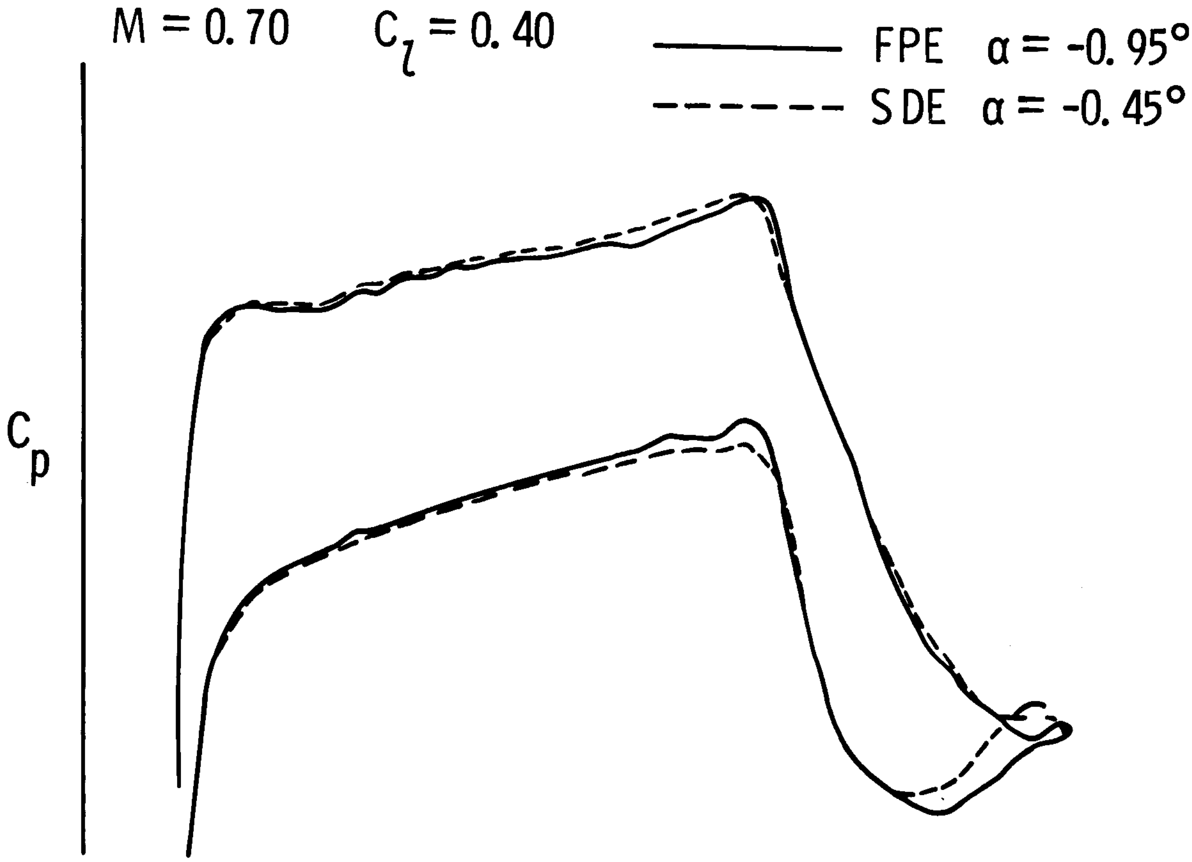
Analyses were made on the subject wing in the three-dimensional WBPPW code and compared to results from the two-dimensional Korn-Garabedian code to assess the applicability of the two-dimensional code. From these results, it was seen that the flow field around the outer panel had two-dimensional behavior implying that the Korn-Garabedian code could be used in the design process.

- Which codes could be used?
- How should the strake be represented?
- How should the body/ nacelle be represented?

APPLICATION OF SMALL-DISTURBANCE CODES TO NLF AIRFOILS

There was also concern about applying the small-disturbance code to the natural laminar flow airfoils because of questions relating to the capability of the code to accurately predict leading-edge pressure distributions. This is essential since pressure distributions conducive to NLF have favorable pressure gradients from the leading edge aft to the transition location. If the leading-edge pressures are inaccurate, potential problems with the design could be masked. To gain confidence, an NLF airfoil was analyzed using the two-dimensional option in the WBPPW code and compared to results from the Korn-Garabedian code.

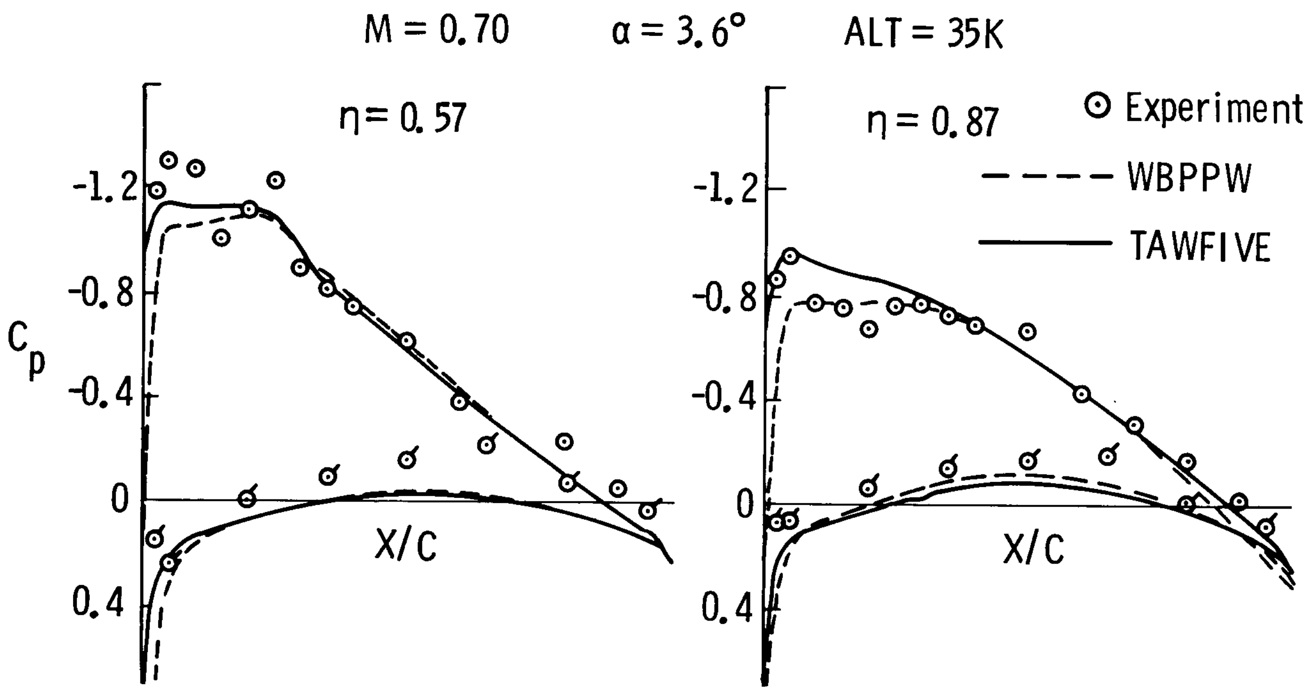
The results showed only minor differences in the pressure distributions, demonstrating that the small-disturbance code could accurately predict the leading-edge pressures of an NLF airfoil.



COMPARISON OF FLIGHT TEST AND COMPUTATIONS

A flight test of the F-14 was conducted to explore the test envelope for the VSTFE and to obtain wing pressure data on the basic aircraft (data to be published). From these data, four flight points were designated to be of primary interest. Three of the points correspond to corners of the flight envelope for the VSTFE, and the remaining point is an intermediate flight condition.

Analyses were made in the WBPPW and TAWFIVE codes at the flight Mach number and measured angle of attack. Overall, the comparisons are quite good. Several observations need to be made concerning the comparisons. First, the flight data showed a flow expansion at the leading-edge followed by a compression that neither code predicted. This indicated that possibly the leading-edge slat deflected under load. Static loading corresponding to flight loads confirmed this. The differences seen in leading-edge expansions between the two codes is consistent with the code formulations. Shock resolution is much better in the WBPPW code results because of the denser grid in that region as compared to the TAWFIVE code. Additionally, the TAWFIVE code uses conservative differencing where WBPPW uses nonconservative differencing, which accounts for the discrepancy in shock location.



DESIGN APPROACH

Based on the comparisons of the potential flow codes with the flight data and the evaluations of the applicability of the two-dimensional and three-dimensional codes to this design problem, it was felt that an integrated two-dimensional/three-dimensional design process could be formulated. The design process that was used was not formulated a priori but evolved with the program. This loosely defined approach is as follows:

> Equivalent sectional lift coefficients were defined as a function of span for the design point (high altitude, M = 0.70) based on the flight test data. Modifications were made to the sectional contours to yield a slightly favorable gradient from the leading-edge to midchord by using two-dimensional analysis and design procedures followed by a three-dimensional design code (FLO-22NM) to reduce any adverse three-dimensional effects.
>
> The resulting sections were then meticulously refaired, smoothed, and analyzed once again with the two-dimensional analysis codes.
>
> Finally, the outer panel was analyzed at the design and off-design conditions with the TAWFIVE and WBPPW codes as part of the complete F-14 configuration.

DESIGN POINT

$M = 0.7 \qquad C_l = 0.60$

- 2–D DESIGN
- 3–D ANALYSIS AND REDESIGN
- 2–D DESIGN, FAIRING, AND SMOOTHING
- 3–D TAWFIVE ANALYSIS

DESIGN CONSTRAINTS

The physical constraints on the modifications evolved with the design program. The final constraints and supporting rationale are:

The upper surface could be modified from the leading-edge to the spoiler hinge line (x/c = 0.60) in order to utilize the spoilers for roll control. Modifications on the lower surface were limited to the first 10-percent chord.

The thickness of the glove at the spoiler hinge line must be less than 1.0 inch. This constraint was imposed to ensure spoiler effectiveness. For reference, the wing mean chord is 105.66 inches.

The thickness of the glove was required to be a minimum of 0.65 inches. This constraint was required to minimize the possibility of the leading-edge slat deflecting under load.

The rolling moment resulting from the asymmetric configuration was required to be less than 0.01 over the flight test envelope.

- UPPER SURFACE MODIFICATION
  0.0 < X/C < 0.60

- LOWER SURFACE MODIFICATION
  0.0 < X/C < 0.10

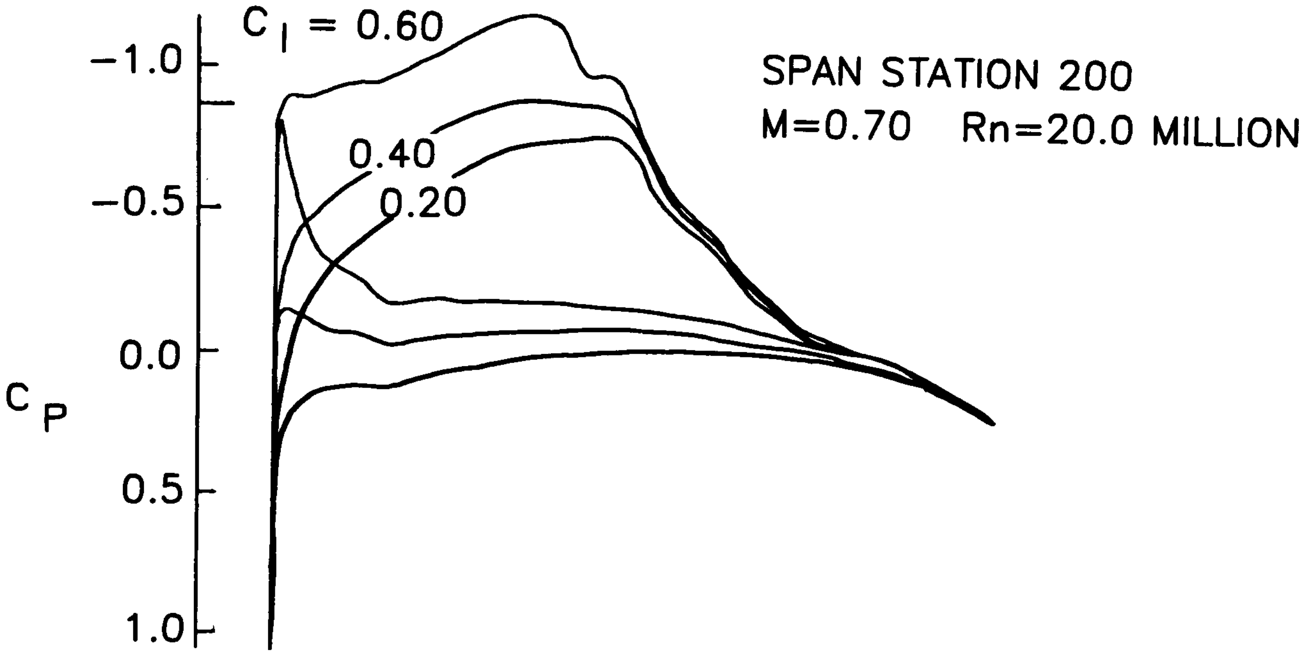
- INCREMENT AT SPOILER HINGE LINE
  LESS THAN 1.0 INCH

- INCREMENT OVER GLOVE REGION A MINIMUM
  OF 0.65 INCHES

- DIFFERENTIAL ROLLING MOMENT LESS THAN 0.01

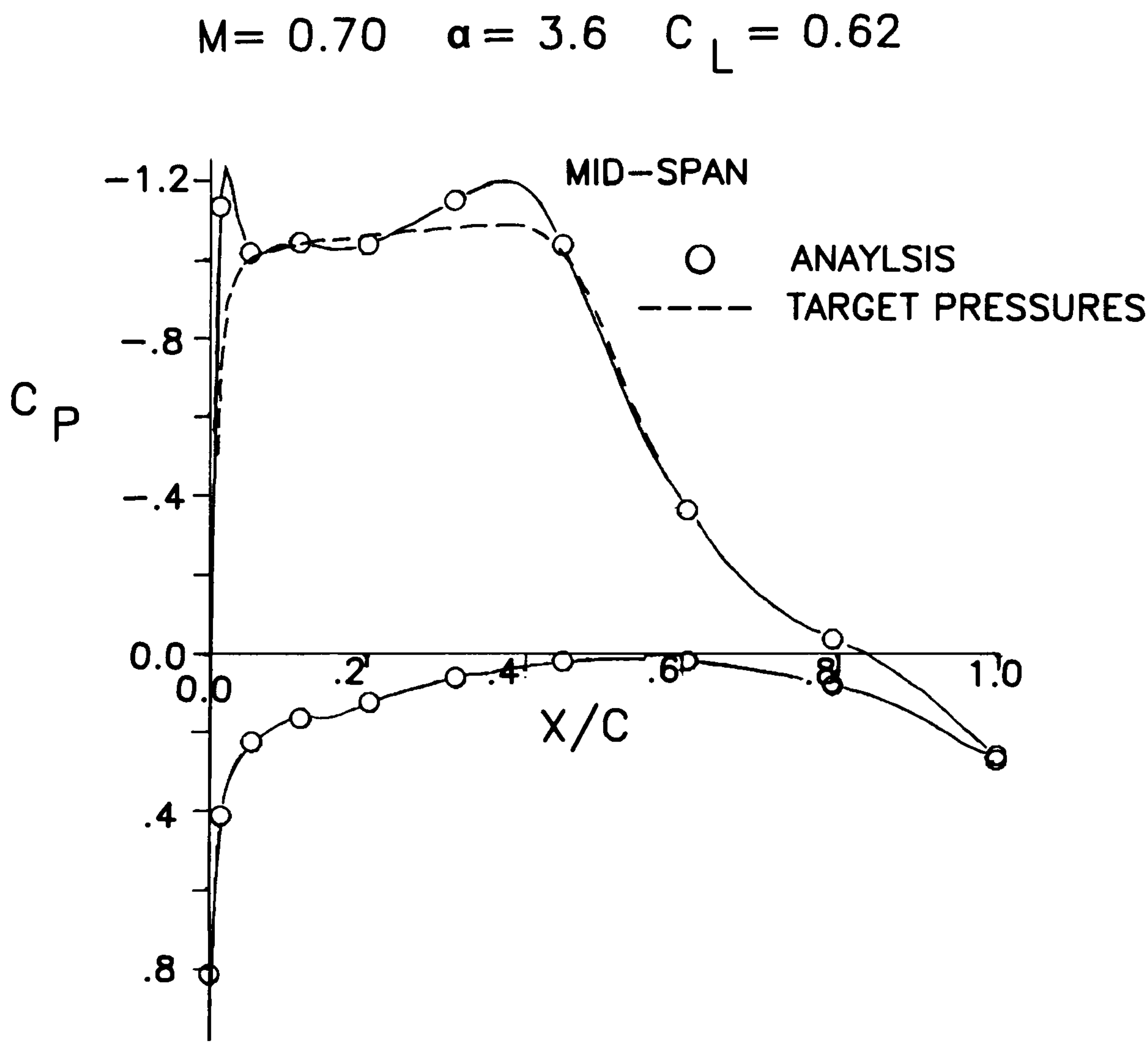
DESIGN AIRFOIL MEETING FINAL CONSTRAINTS

The design point selected corresponded to a "worst case" condition for the targeted Mach number (M = 0.70). This condition corresponded to the highest altitude, hence the largest lift coefficient for 1-g flight. If the sectional contours could be modified such that a slightly favorable gradient could be generated from the leading-edge to the midchord region at this condition, then at lower altitudes there would be an even more favorable pressure gradient. Five defining stations were chosen to be recontoured using linear lofting between defining stations. These corresponded to the inboard and outboard extent of the glove and three intermediate defining stations. With two-dimensional analysis and design procedures, upper surface contours were defined which met the aerodynamic and physical constraints for each defining station. A favorable pressure gradient was observed for a range of lift coefficients on the design airfoil as well as a favorable pressure gradient aft of the leading-edge to the pressure rise.



3-D ANALYSIS AND DESIGN

The next step in the design process was to analyze the recontoured outer panel in a three-dimensional environment and further modify the configuration to eliminate any adverse three-dimensional effects. For this phase, a derivative of FLO-22 was used because of its capability to modify a wing contour to yield a desired pressure distribution (ref. 5). The results of a wing-alone analysis revealed two undesirable characteristics in the pressure distributions. A pressure peak at the leading edge appeared as did a flow expansion ahead of the shock. Target pressure distributions were generated to minimize the effects of these two characteristics. A design run was made using the target pressures, which yielded new wing section contours. The design algorithm handled the leading edge quite well; however, the contour changes near the shock caused pressure oscillations. A two-dimensional design code was brought into play that systematically smooths the airfoil curvature while attempting to generate a specified target pressure distribution. Final contours were produced using this code that met the design constraints at each defining station.



FINAL DESIGN AIRFOIL

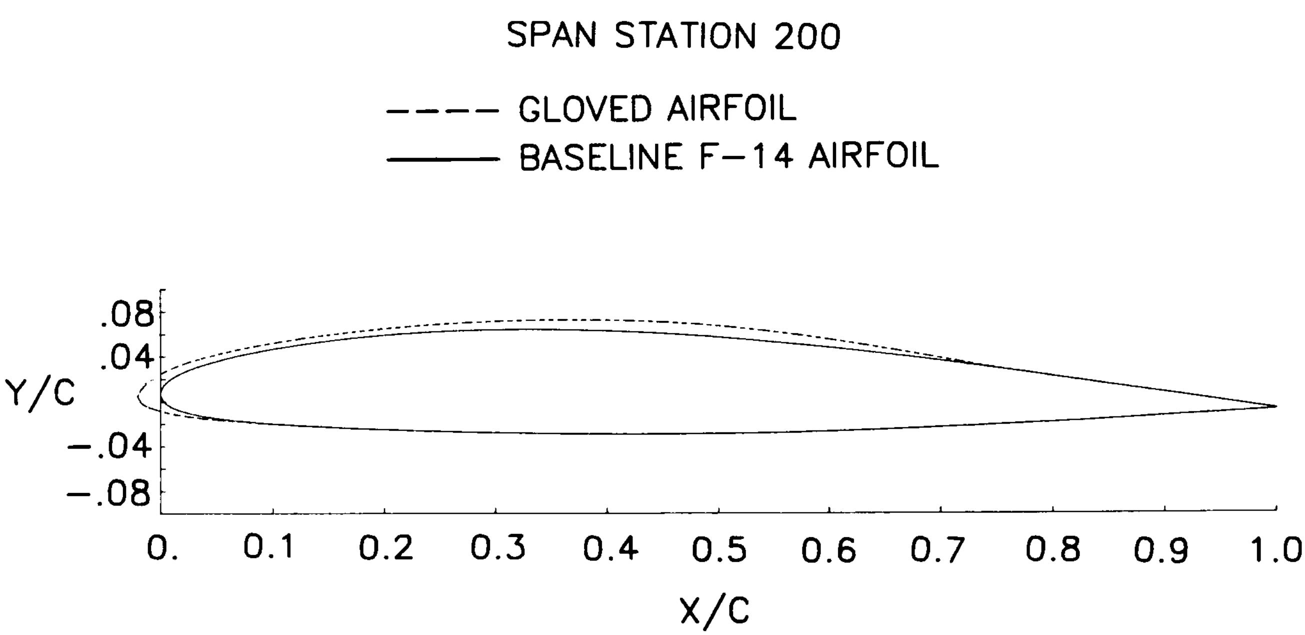
The final design airfoils at each defining station met these geometric constraints:

Upper surface modifications were made from the leading-edge to the spoiler hinge line. Lower surface modifications were made on the first 10-percent chord.

Maximum thickness at the spoiler hinge line was less than 1.0 inch. Minimum thickness of the contour over the entire gloved surface was at least 0.65 inches.

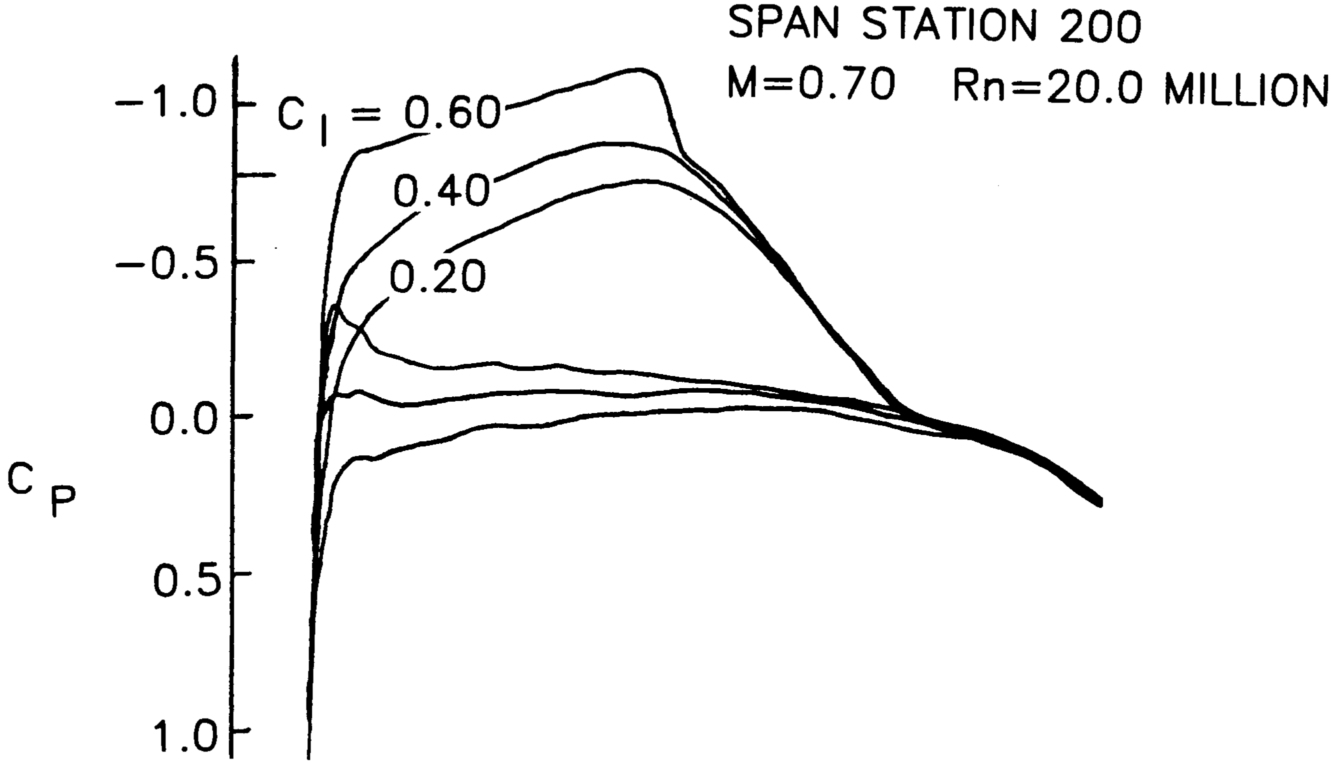
An incremental rolling moment between the NASA Langley glove and the Boeing glove was less than 0.01.

Instrumentation leads were to be routed inside the leading edge of the glove, hence it was necessary to extend the glove leading edge 2.0 inches in front of the actual leading-edge of the wing.



FINAL NLF F-14 AIRFOIL DESIGN

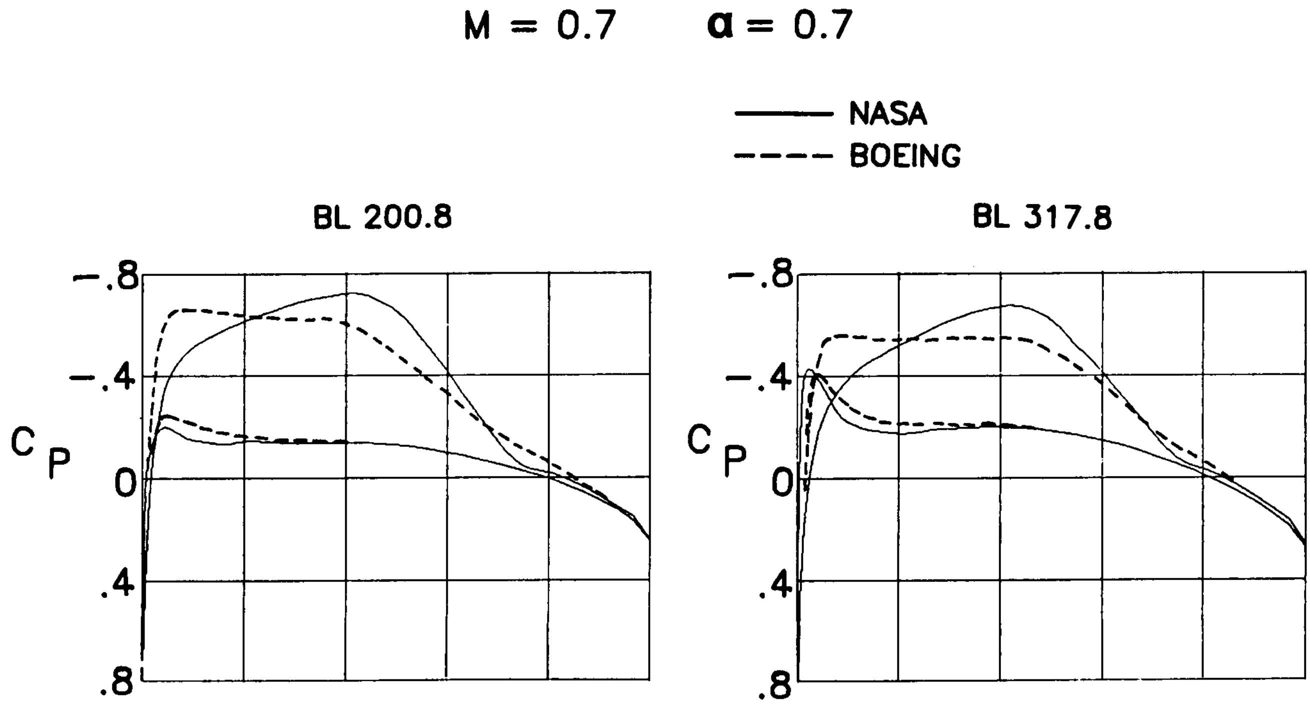
Two-dimensional analysis of the final airfoil design for each defining station show the desired favorable gradient from the leading-edge to the midchord region over a range of section lift coefficients at M = 0.70. Results gave no evidence of adverse effects from the final section contour modifications. Pressure distributions for the midspan defining station at M = 0.70 for various sectional lift coefficients are shown and are representative of the pressure distributions at the other defining stations.



3-D ANALYSIS OF GLOVE DESIGNS

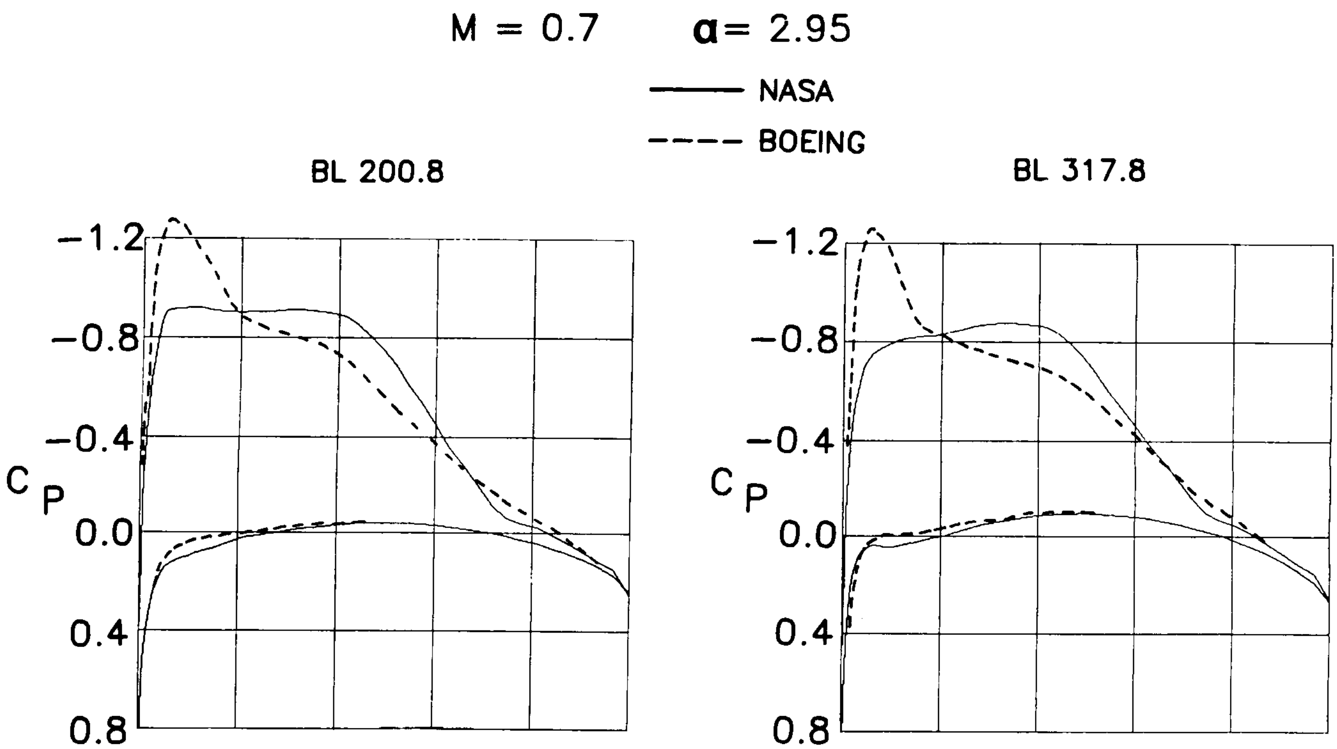
Final computational verification of the design was realized by analyzing the entire configuration (fuselage, nacelles, strake, and outer panel) in the TAWFIVE code. Results show that the design objectives were met over the range of lift coefficients corresponding to the altitudes of interest at M = 0.70. In addition, an analysis solution was obtained at an off-design point which corresponds to Boeing's design point, M = 0.80. The following three figures reflect those results.

The first figure shows the NASA glove design as compared to the Boeing glove design at the low altitude (25,000 feet, C_L = 0.35) case at M = 0.70. Note that this condition is an off-design condition for the Boeing design; however, the data for the Boeing glove are included since the designs will be flown simultaneously.



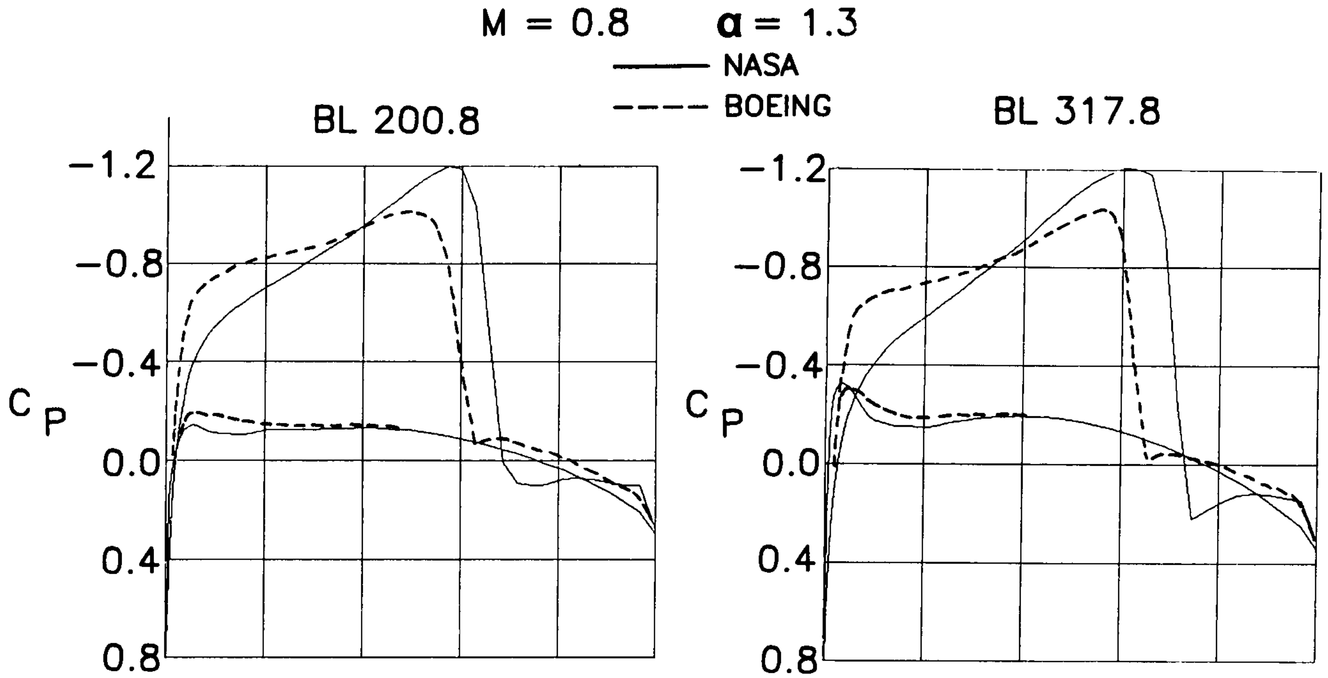
3-D ANALYSIS OF GLOVE DESIGNS

Presented in this figure are the results for the high altitude (35,000 feet, C_L = 0.52) case at the targeted Mach number, M = 0.70. This condition corresponds to the "worst case" at the targeted Mach number. The results show a slightly favorable gradient from the leading edge aft to the pressure rise. Again, this is an off-design condition for the Boeing glove.



3-D ANALYSIS OF GLOVE DESIGNS

At the high altitude (35,000 feet) off-design Mach number for the NASA glove, M = 0.80, the boundary-layer analysis gave no indication of flow separation. Since the computational analysis predicted acceptable results and the design constraints were met, the glove design was frozen at this point. Note that this off-design point for the NASA glove design is the "worst case" design condition for the Boeing glove design.



STATUS OF F-14 VSTFE

Glove design has been completed for the VSTFE and model fabrication is underway for a wind-tunnel test to be conducted in the NTF during the early summer 1985. The objectives of the test are to obtain data to verify the glove design and safety-of-flight data for support of the flight-test program. Flight-test instrumentation techniques will be validated in a program which will be flown in late summer or early fall 1985. A "cleanup" glove has been fabricated for the F-14 outer panel which employs the physical constraints described previously and corresponds to the baseline F-14 outer panel contour. Any manufacturing or instrumentation problems uncovered during this program can be addressed before the NLF glove experiment is flown. Manufacture of the NLF glove will commence in the last quarter of 1985 with the flight test following 9 to 12 months later.

- COMPUTATIONAL DESIGN COMPLETED
- WIND TUNNEL TEST FOR DESIGN VERIFICATION AND SAFETY OF FLIGHT – NTF, JUNE 1985
- FLIGHT TEST GLOVE DESIGN – DRYDEN FLIGHT TEST FACILITY, SUMMER 1986

3-D DESIGN OF A HIGH-ASPECT-RATIO NLF TRANSONIC WING

The second design configuration reported in this study is a high-aspect-ratio, natural laminar flow wing for a corporate transport. The objective for this program was to design a wing which would operate efficiently at a transonic cruise point and generate significant runs of laminar flow on both upper and lower surfaces. Because the aircraft had a single jet engine, it had to meet a landing speed requirement of 68 knots. This requirement dictated a relatively large wing area and a maximum lift coefficient of about 3.8. The large wing area, however, meant that at the cruise Mach number of 0.7 the lift coefficient was only 0.25. Other design constraints included: 1) a maximum thickness-to-chord ratio of at least 12.5 percent (for fuel volume and landing gear storage) and 2) good aerodynamic characteristics with a fully turbulent boundary layer.

DESIGN A WING WHICH OPERATES EFFICIENTLY AT A TRANSONIC CRUISE DESIGN POINT AND WHICH GENERATES SIGNIFICANT RUNS OF LAMINAR FLOW ON BOTH UPPER AND LOWER SURFACES

CONSTRAINTS AND DESIGN CONDITIONS :

- CRUISE AT M=0.70 AND $C_L = 0.25$
- T/C > 12.5%
- $C_{l_{MAX}} = 3.8$
- GOOD AERODYNAMIC CHARACTERISTICS WITH A FULLY TURBULENT BOUNDARY LAYER

APPROACH

The approach used in the design of this wing was to apply both two-dimensional and three-dimensional transonic potential flow methods which have been validated for transport configurations. The major airfoil design modifications were made using the Garabedian and Korn two-dimensional analysis code. This code has been widely used by industry and is robust and accurate. The two-dimensional subsonic panel code was used to evaluate the low speed, high lift characteristics of the final airfoil. Analyses of the configuration with the airfoils developed from the two-dimensional design methods were made using two of the three-dimensional transonic codes described earlier. The primary code used was TAWFIVE, a full-potential wing/fuselage code which includes a three-dimensional boundary layer calculation to model viscous effects. This code has not been extensively validated with data, but has shown some very encouraging results. A second code, WBPPW, which has been run for many configurations, was also used to add confidence to the TAWFIVE results.

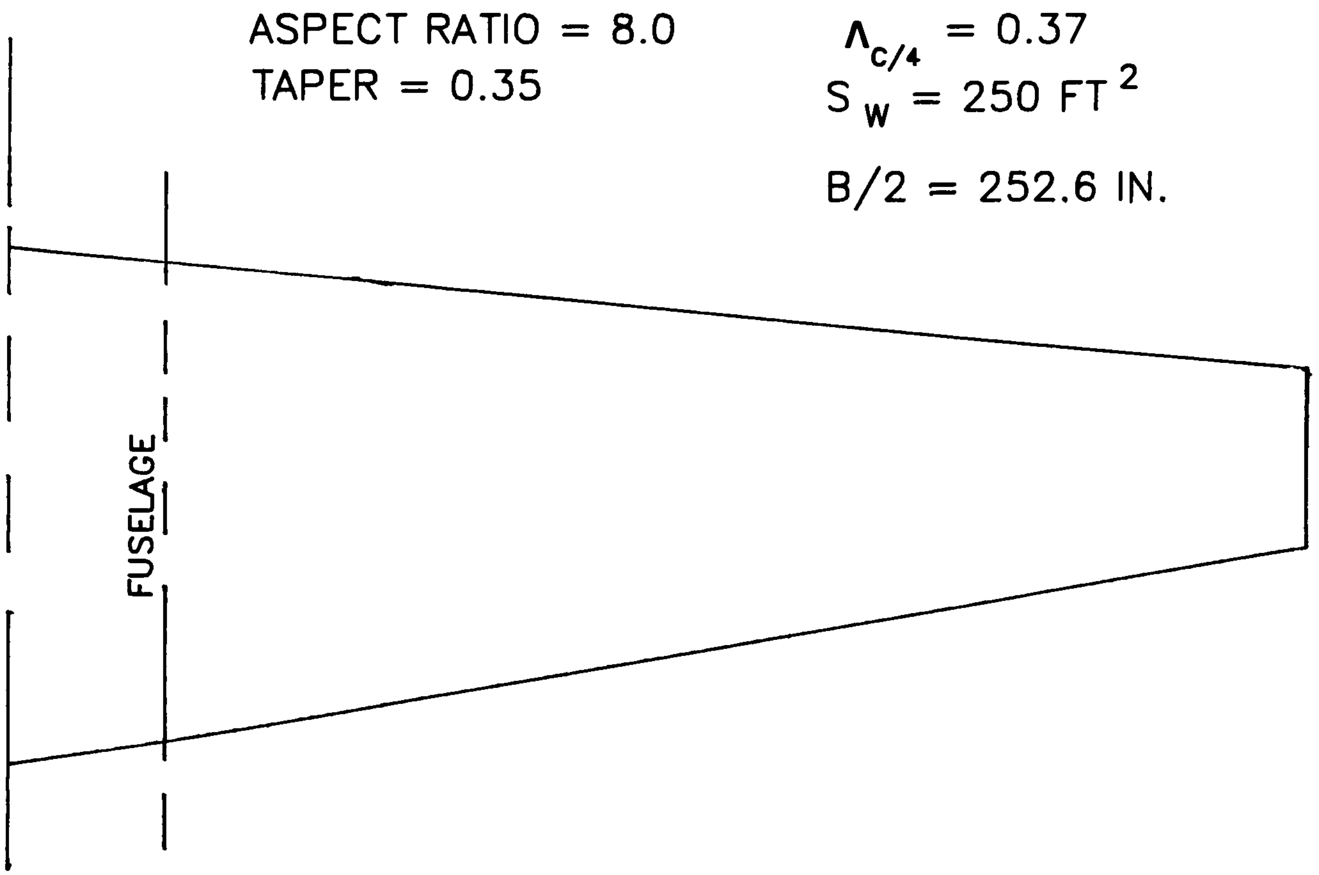
- APPLY RELIABLE 2-D AND 3-D TRANSONIC POTENTIAL FLOW METHODS WHICH HAVE BEEN VALIDATED FOR TRANSPORT APPLICATION.

- TWO-DIMENSIONAL ANALYSIS:
  - **NYU GARABEDIAN AND KORN**
  - **HIGH LIFT CODE**

- THREE-DIMENSIONAL ANALYSIS:
  - **TAWFIVE (FPE/ 3-D BL)**
  - **WBPPW (ESD/ 2-D BL)**

WING PLANFORM

The design variables used in this study were airfoil shape and wing twist. The wing planform was specified a priori and had a span of 252.6 inches, an area of 250.0 square feet, an aspect ratio of 8.0, and a taper ratio of 0.35. The essentially unswept quarter-chord line reduces the chances that crossflow instabilities in the laminar boundary layer would cause premature transition. The low sweep and high aspect ratio of the wing meant that the two-dimensional airfoil design pressures were maintained on the three-dimensional wing except for the regions near the tip and the side of the fuselage.

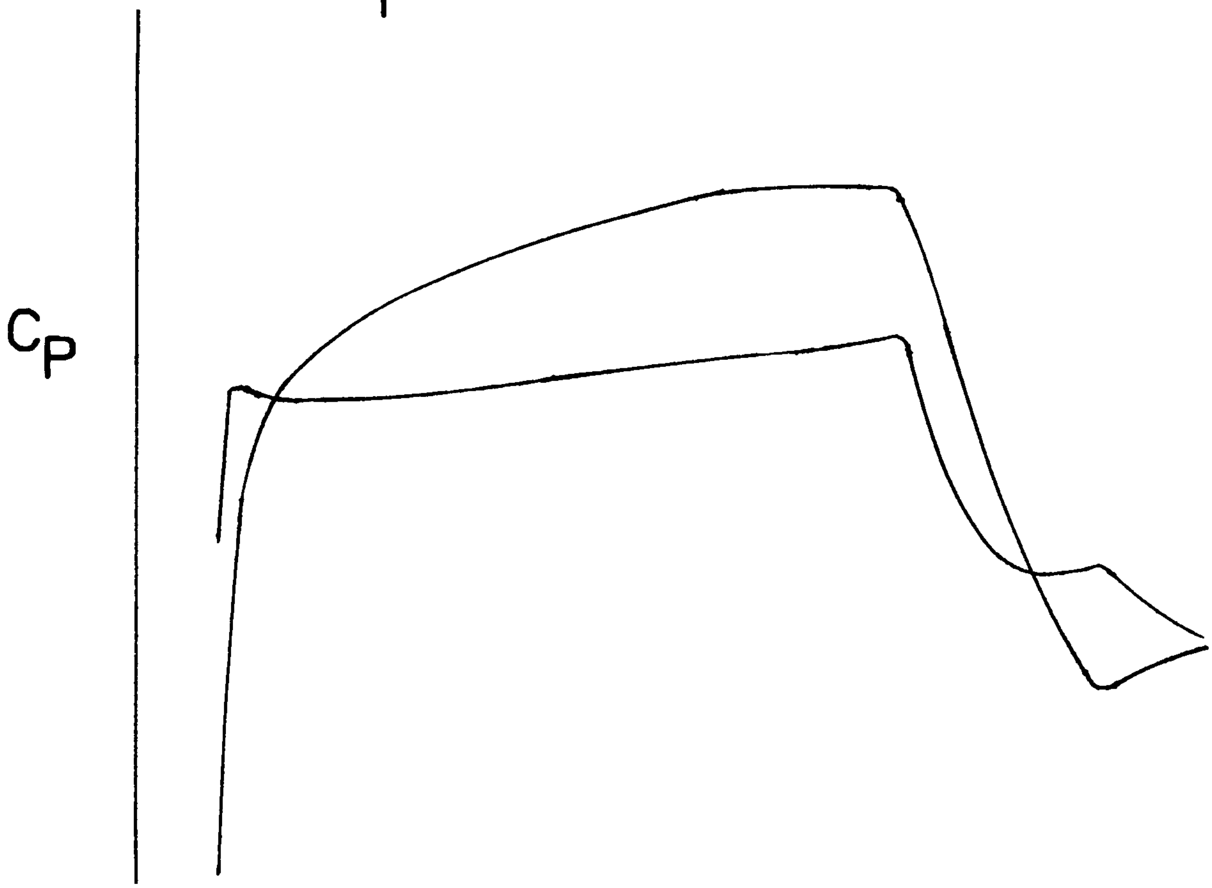


INITIAL DESIGN

The starting point for the airfoil design was a 14-percent thick medium-speed NLF airfoil developed at NASA Langley by Viken, reference 9. This airfoil maintained laminar flow back to about 70-percent chord on both surfaces at a Mach number of 0.4, a lift coefficient of 0.4, and a Reynolds number of 10 million. The first design modification involved scaling down the airfoil thickness to account for the increase in design Mach number. The trailing edge camber was also reduced to lower the design lift coefficient to 0.25. The pressure distribution at the design Mach number of 0.7 and lift coefficient of 0.25 was calculated using the two-dimensional code and is shown below.

Two undesirable features were present in these results. First, the slight pressure peak at the leading edge of the lower surface could cause the boundary layer on that surface to transition. Second, the adverse gradient that begins at about 70-percent chord on each surface is too steep and would probably cause the flow to separate.

$M = 0.70 \quad C_l = 0.25 \quad Rn = 11. \times 10^6$

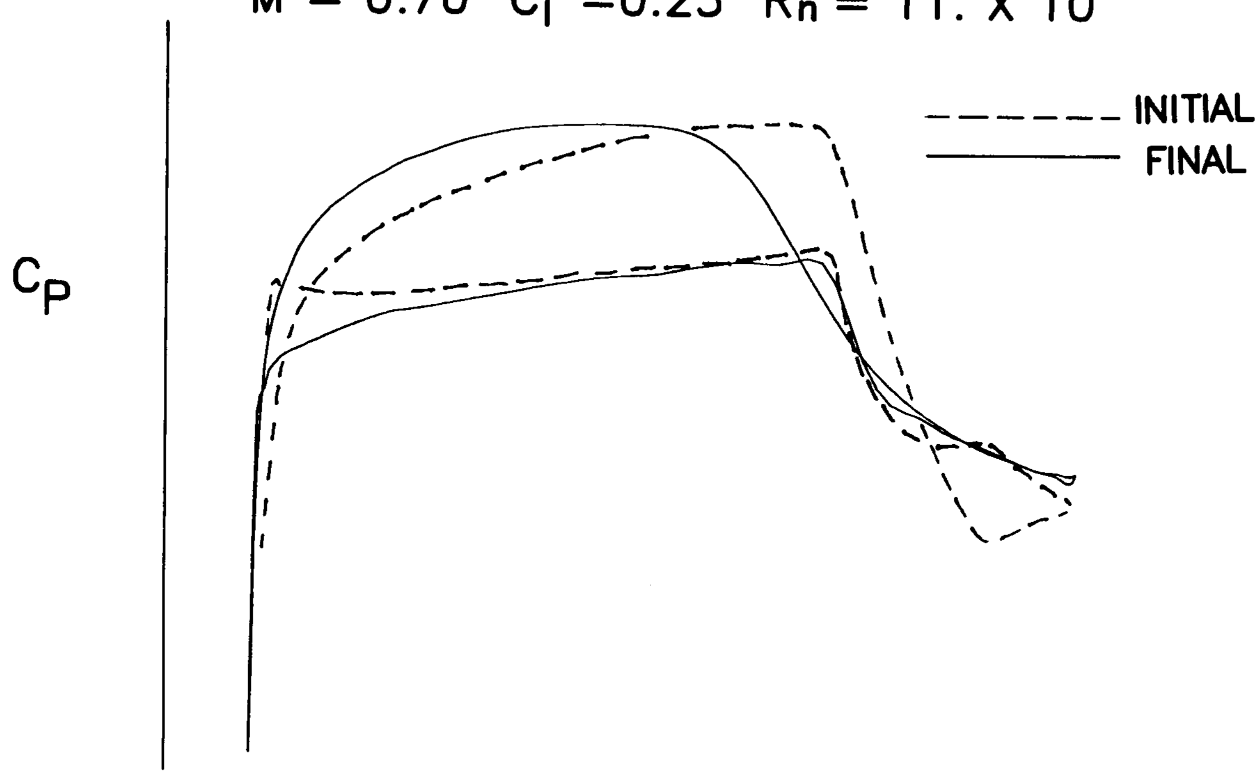


COMBINATION AIRFOIL DESIGN

The initial design airfoil was modified to eliminate the undesirable features in its pressure distribution. The modified shape was evaluated using both two-dimensional and three-dimensional analysis codes. The pressure distributions calculated by the two-dimensional Garabedian and Korn code for both the initial and final airfoil designs are shown in the accompanying figure. The leading-edge peak on the lower surface was eliminated and the adverse pressure gradients were softened on both surfaces. This reduction in the adverse gradients eliminated the flow separation on the upper surface; it also, however, limited the extent of laminar flow to about 50-percent chord on that surface. It is interesting to note that almost no load is carried on the last 30-percent of the airfoil.

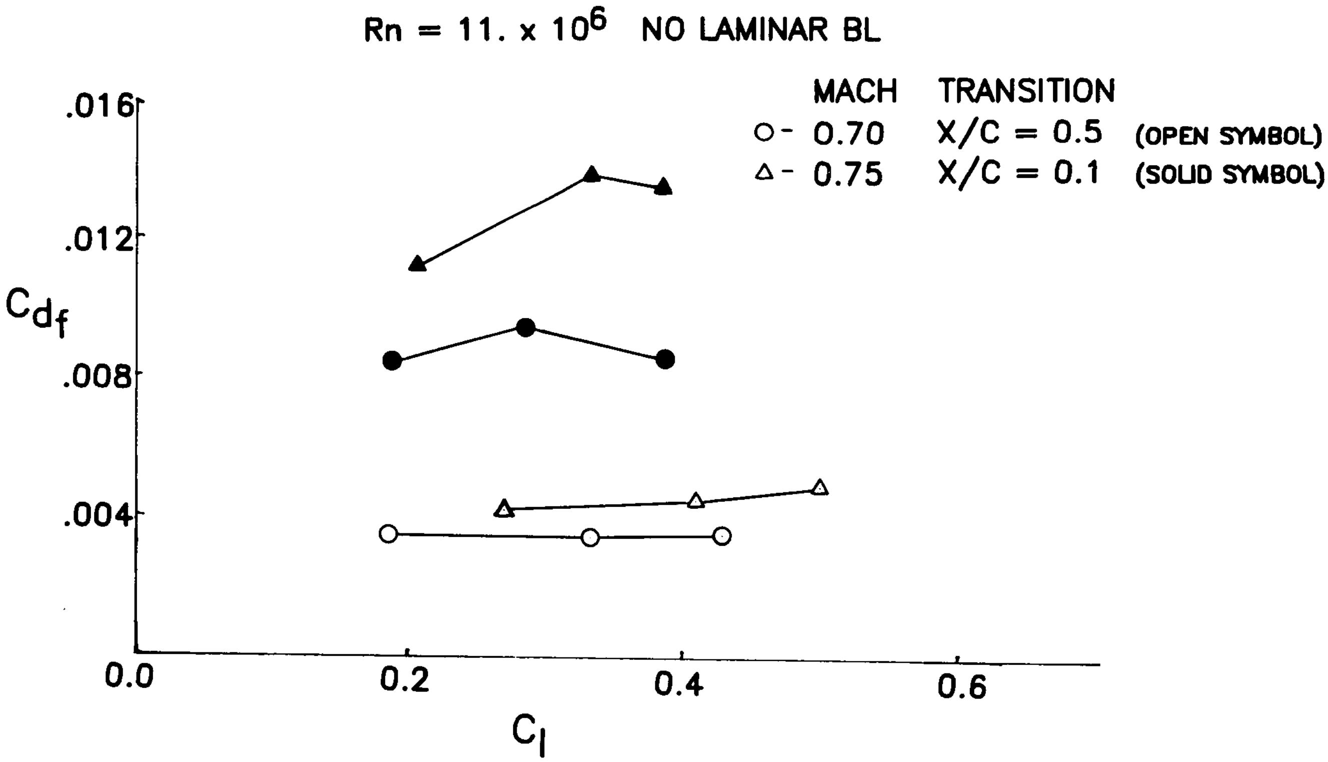
HYBRID AIRFOIL DEFINED WITH VIKEN DESIGNED UPPER SURFACE AND CAMPBELL DESIGNED LOWER SURFACE LEADING EDGE

$M = 0.70 \quad C_l = 0.25 \quad R_n = 11. \times 10^6$



VARIATION OF SKIN FRICTION DRAG COEFFICIENT WITH SECTIONAL LIFT COEFFICIENT

Since the primary reason for using NLF airfoils is a reduction in viscous drag, an estimate of this drag benefit was made using the Garabedian and Korn program. Runs were made at Mach numbers of 0.70 and 0.75 and lift coefficients from about 0.2 to 0.4. Because the code only has a turbulent boundary-layer model, the laminar flow case was simulated by beginning the boundary-layer calculation at an assumed transition location of 50-percent chord. Transition for the turbulent flow case was fixed at 10-percent chord. As seen in this figure, a reduction of about 50 counts of drag is obtained at a Mach number of 0.70 (circles) when going from the turbulent case (solid symbols) to the laminar flow case (open symbols). An even greater reduction of about 80 counts occurs at a Mach number of 0.75. Again, these estimates are a little high since the laminar flow region was modeled as contributing no drag, but should be representative of relative reductions in turbulent boundary layer skin friction drag.

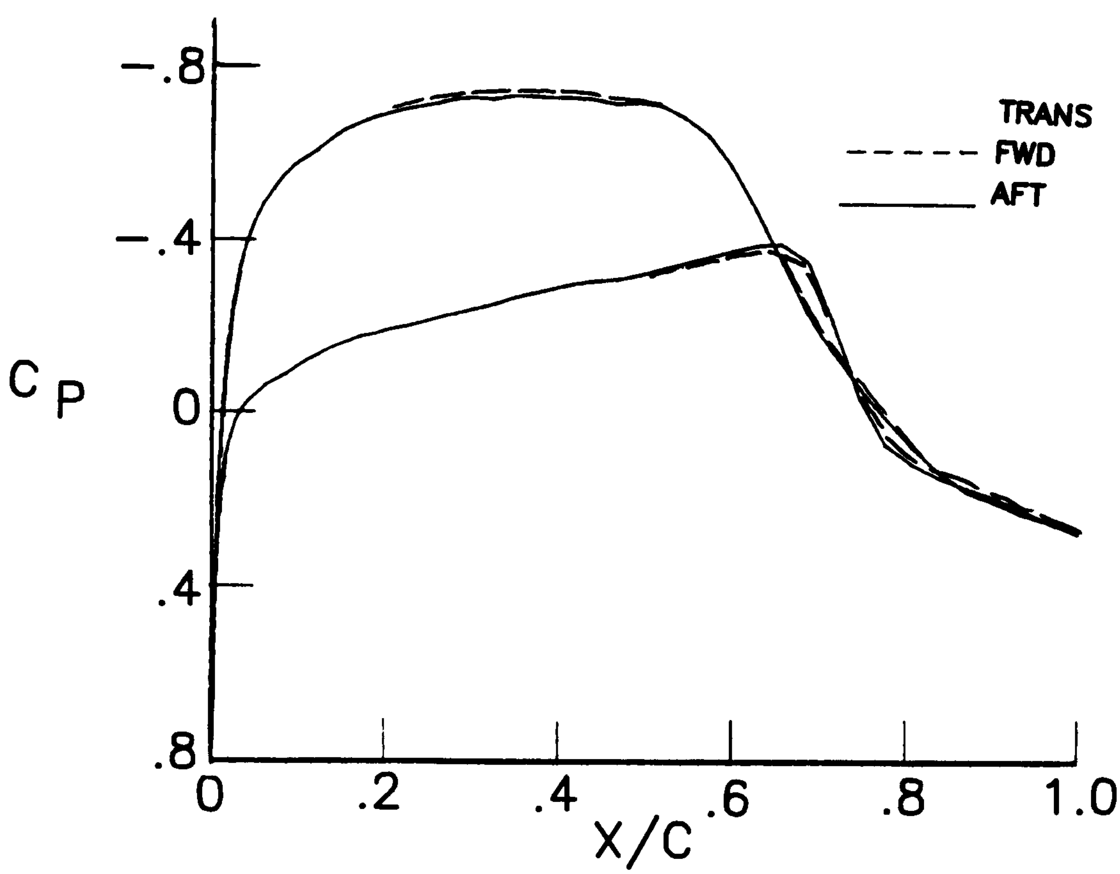


3-D EVALUATION OF COMBINATION AIRFOIL

One of the design requirements for this configuration was that there be no large changes in the aerodynamic characteristics, except for an increase in viscous drag, if laminar flow was lost. In order to examine this condition, the wing/body configuration using the final airfoil design was run in TAWFIVE with transition set at forward and aft positions. For the laminar flow case, transition was set at 50-percent chord on the upper surface and 65-percent chord on the lower surface; the turbulent flow case had transition fixed at the leading edge. The TAWFIVE code, unlike the Garabedian and Korn two-dimensional code, did have a laminar as well as a turbulent boundary-layer model. The pressure distributions calculated for a station near mid-semispan show that transition location has very little effect for these cases. This was not surprising; however, it was important that no separation was predicted for either case. This means that if the wing suddenly lost laminar flow while in a cruise condition, no sudden change in aircraft flying qualities would occur.

PLANFORM 2 NON-LINEAR TWIST

$C_L = 0.25 \quad M = 0.7 \quad \eta = 0.383$



COMPUTATIONAL WING DESIGN EFFORT

This wing design study covered several areas in addition to the design of the airfoil sections which are not described in detail in this report. A summary of the design effort is given in this chart. The study began with the definition of initial airfoil sections. After evaluating them using a two-dimensional airfoil code, an initial three-dimensional analysis was made. This run identified some problem areas and led to two parallel airfoil modification efforts. While the airfoil modifications were being made using two-dimensional codes, the three-dimensional codes were being utilized to predict the effects of a proposed planform modification. Also, an investigation of several linear and non-linear twist distributions was carried out. The two airfoils developed in the parallel efforts were combined to form a hybrid airfoil that retained the good characteristics of each one at the design conditions. This airfoil was then evaluated at numerous off-design conditions using both two-dimensional and three-dimensional analyses.

One additional area of study was the design of a leading-edge extension for the outboard part of the wing. This type of wing modification has been found to be very effective in improving the stall departure and spin resistance characteristics of a general aviation aircraft, reference 10. Two extensions were designed for this aircraft, both extending from about 75-percent semispan to the tip. The leading edges of the airfoils were extended forward 2- and 3-percent chord and drooped for the two cases. Both two-dimensional and three-dimensional transonic codes were used to evaluate the modified airfoils at the cruise condition and the low-speed, high-lift chararcteristics were predicted using the two-dimensional subsonic code.

- INITIAL 3-D ANALYSIS
- PROBLEM IDENTIFICATION AND AIRFOIL MODIFICATION
- PLANFORM MODIFICATION
- NON-LINEAR TWIST INVESTIGATION
- HYBRID AIRFOIL DEFINITION
- OFF-DESIGN ANALYSIS
- LEADING EDGE EXTENSION DESIGN

SUMMARY

In summary, state-of-the-art computational methods were used to design a high aspect ratio NLF wing for a corporate transport. This effort included making a total of about 80 three-dimensional analysis runs and dozens of two-dimensional analysis runs over a period of about two months. The final design was predicted to maintain laminar flow back to 50-percent chord on the upper surface and 60-percent chord on the lower surface at the design conditions (Mach number of 0.7 and lift coefficient of 0.25). The requirement of no boundary-layer separation if transition should occur at the leading edge was also met for the design point. Based on two-dimensional calculations, a drag divergence Mach number of 0.75 was determined, which would give an adequate margin above cruise.

One purpose of this study was to evaluate the use of these codes in a project environment. Strengths and weaknesses of the various codes were identified and new computational tools were developed to complement the two-dimensional and three-dimensional design methods.

- COMPUTATIONAL NLF WING DESIGN COMPLETED

- APPROXIMATELY 65 3-D ANALYSIS RUNS TAWFIVE
  APPROXIMATELY 15 3-D ANALYSIS RUNS WBPPW
  DOZENS OF 2-D ANALYSIS RUNS

- 50%/60% LAMINAR FLOW ON US/LS AT DESIGN POINT

- NO SEPARATION WITH TRANSITION AT LEADING EDGE AT DESIGN POINT

- DRAG DIVERGENCE MACH = 0.75 AT CRUISE

- IDENTIFIED INADEQUACIES AND DEVELOPED METHODS TO COMPLEMENT 2-D AND 3-D DESIGN EFFORTS

CONCLUDING REMARKS

In conclusion, computational wing design methodologies were successfully applied in two unique programs. The two-dimensional and three-dimensional aerodynamic codes used in these studies proved to be robust and reliable in a stringent schedule environment. The automated design procedure available in one of the three-dimensional codes yielded excellent results and the inclusion of that procedure or a similar one in the other aerodynamic codes is being pursued. Some deficiencies in the capabilities of the codes were identified and possible corrections and improved running strategies are being addressed. The final accuracy of the design methods will be evaluated when wind-tunnel tests of both configurations are completed.

- SUCCESSFULLY APPLIED COMPUTATIONAL WING DESIGN METHODOLOGIES IN TWO UNIQUE PROGRAMS
- 2–D AND 3–D CODES WERE ROBUST AND RELIABLE IN STRINGENT SCHEDULE ENVIRONMENT
- AUTOMATED DESIGN PROCEDURES YIELDED EXCELLENT RESULTS
- DEFICIENCIES IN CAPABILITIES WERE IDENTIFIED AND ARE BEING ADDRESSED
- WIND TUNNEL DATA WILL VALIDATE THE COMPUTATIONAL WING DESIGNS

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