DEVELOPMENT OF INVERSE INVISCID

TRANSONIC SOLUTION METHODS

(NASA-CR-142016) DEVELOPMENT OF INVERSE INVISCID TRANSONIC SOLUTION METHODS Final Report, Nov. 1973 - Dec. 1974 (Texas A&M Univ.) 23 p HC $3.25 CSCL 01A

N75-15610

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G3/02 07767

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(NASA Grant No. NGR-44-001-157)

TAMRF Report No. -3033-7403

TEXAS ENGINEERING EXPERIMENT STATION

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The NASA Technical Officer for this grant is Mr. J. C. South, Jr., Theoretical Aerodynamics Branch, NASA Langley.

I. Introduction

During the past 13 months considerable progress has been made under the present grant. The application of time-like damping and Riegel's Rule to the transonic small perturbation equations has been investigated and reported. In addition, a computer program that utilizes the full equations in a rectangular coordinate system has been developed. This program uses time-like damping and can be used in either the direct (body specified, surface pressure and flowfield unknown) or design (surface pressure specified, body and flowfield unknown) mode. A brief discussion of the results of these efforts is presented in the following sections.

II. Small Perturbation Work

The nonlinear transonic small perturbation equations have been analyzed according to the stability criteria presented by Jameson, and it has been determined that time-like damping is necessary in order to ensure stability. This damping, which is of the form ϕ_{xt}, has been incorporated into the existing small perturbation equation computer program. Typical results are shown on Figure 1. For this case, a NACA 0006 at Mach 0.9, good results could only be obtained with the inclusion of damping. Similar results have been obtained for NACA 0012 airfoils and lifting cases.

The small perturbation studies have also included the application Riegel's Rule to round-nosed airfoils. In general, the application of Riegel's Rule has been found to improve the results obtainable from the small perturbation equations. This work has been reported. (See list of publications.)

III. Full Equation Work

A summary of this work is included as Appendix A.

IV. Personnel

The following individuals have participated in the project:

Dr. Leland A. Carlson, Associate Professor of Aerospace Engineering

Mr. Don Herron, Senior, Aerospace Engineering

Mr. Raymond Luh, Junior, Aerospace Engineering

V. Publications

The following have been partially supported by this grant:

1. Herron, D. L., "The Application of Riegel's Rule and Time-Like Damping to Transonic Flow Calculations," Presented at 22nd AIAA Southwestern Student Paper Competition, Arlington, TX., April 19th-20th, 1974, also TEES Report No. 3033-7401.
2. Carlson, L. A., "Inverse Transonic Flow Calculations Using Experimental Pressure Distributions," AIAA Journal, Vol. 12, No. 4, pp. 571-572, April 1974.
3. Carlson, L. A., "Transonic Airfoil Flowfield Analysis Using Cartesian Coordinates," NASA-CR-(To be issued).
4. Carlson, L. A., "Transonic Airfoil Design Using Cartesian Coordinates," NASA-CR-(To be issued).

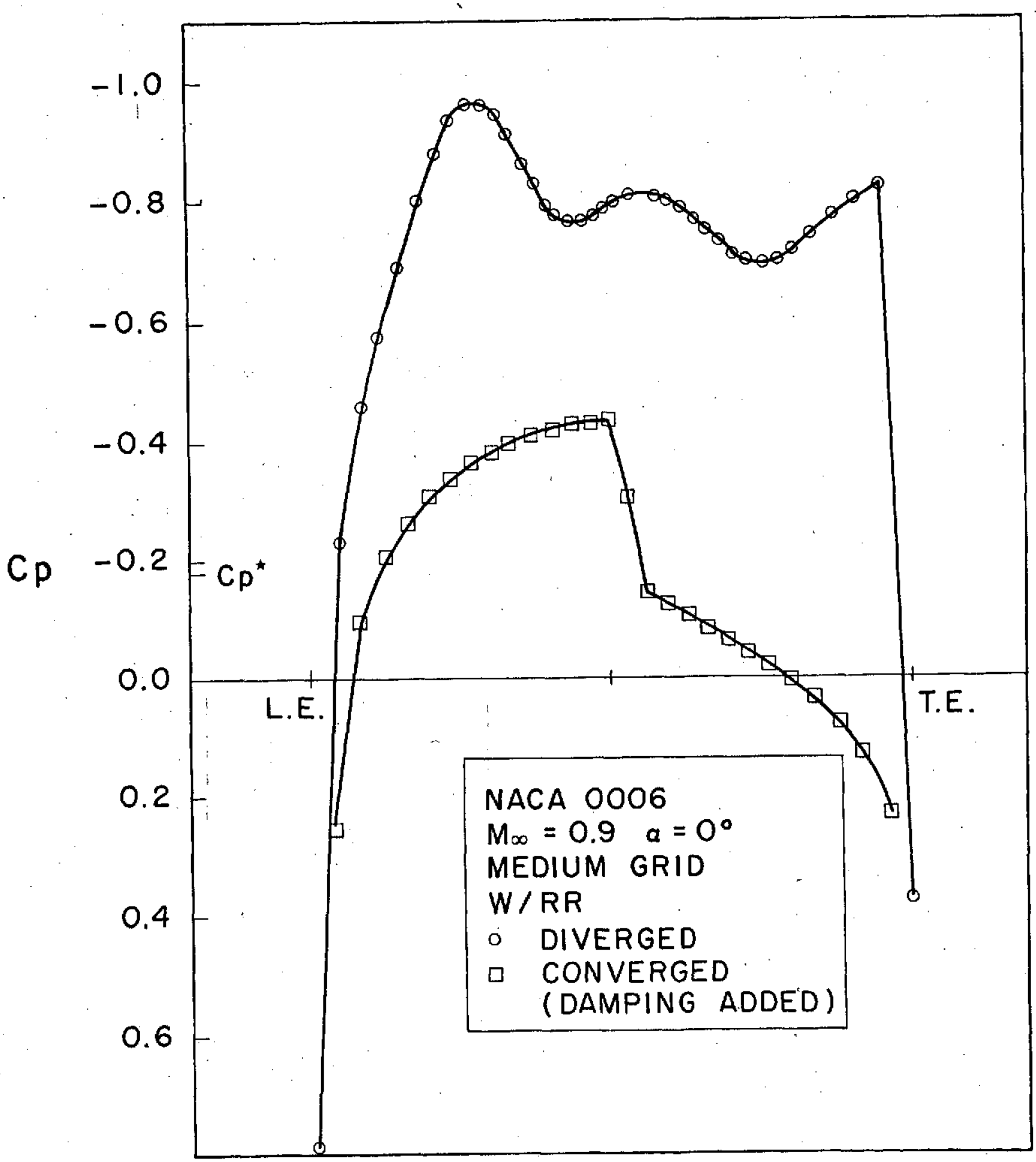

FIGURE 1

THE EFFECT OF TIME-LIKE DAMPING IN THE SMALL PERTURBATION EQUATIONS ON NACA 0012 RESULTS (WITH RIEGEL'S RULE)

Appendix A

TRANSONIC AIRFOIL ANALYSIS AND DESIGN USING CARTESIAN COORDINATES

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December 1974

(Submitted to the 2nd AIAA Computational
Fluid Dynamics Conference, June 19-20, 1975)

ABSTRACT

A numerical method for the design of transonic airfoils and the analysis of the flow about them should not only be accurate but also be as simple as possible in concept and approach. In particular, it should use coordinate systems, input variables, and boundary condition treatments that can be easily understood by the user. In addition, it would be desirable if the method yielded the airfoil design shape for a given set of conditions without iteration and used or computed nose and tail shapes that are aerodynamically and structurally reasonable. Finally, it should not be limited to shocked or shockless flows, but should be able to handle both types.

Previous design methods and programs have either been limited to shockless designs [1] having complicated inputs not easily related to the problem by the user, or have used the small perturbation equations which may be inaccurate for thick blunt-nosed airfoil designs, [2] or have required iterative changes in the desired pressure distribution. [3] The purpose of this paper is to present and discuss a new numerical method suitable for the analysis or design of supercritical transonic airfoils.

In order to achieve accuracy, the method utilizes the full inviscid potential flow equations; and, in order to remain simple it solves the problem in a stretched Cartesian grid system. See Fig. 1. No complicated mappings etc. of the airfoil to a circle or other shape are used. The resultant working computer program has several unique features. First, it can be used in either the direct (analysis) mode in which the airfoil shape is prescribed and the flowfield and surface pressures are determined, or in the inverse (design) mode in which the surface pressures are given and the airfoil shape and flowfield are computed. Second, it uses for the first time in a design program

the rotated finite difference scheme, proposed by South and Jameson,[4,5] which always has the correct zone of dependence in supersonic regions but does not require the coordinate system to be closely aligned to the flow direction. Third, unlike previous methods, the present program determines the airfoil shape simultaneously with the flowfield relaxation solution. Thus, when the converged solution is achieved, the final airfoil design is known, and iteration is not required.

With respect to the rotated difference scheme, it should be noted that the present approach is different from that used in Ref. (4-5). While still rotating the difference scheme and viewing the relaxation process as a time-like procedure, time terms in the streamwise direction are not introduced implicitly as consequence of the manner in which the difference expressions are formulated. Instead they are added explicitly and as in Ref. (4-5) used to control the stability and convergence of the relaxation process. (Note that these time-like terms correspond to the change between relaxation cycles and thus approach zero as the solution converges.) By explicitly adding the time-like damping, no additional damping is required; and the amount of damping required can be easily determined by the user from the maximum local Mach number. In the design case, the latter would be known from the assumed surface pressure distribution. A detailed discussion of the numerical scheme and its stability will be presented in the full paper.

In the Cartesian like system, the airfoil surface and grids lines do not coincide. Since in the design case the surface is not known _a priori_, this lack is not a particular disadvantage, but nevertheless appropriate boundary values at the computational boundary must be determined. The paper discusses various approaches, points out which are unstable and which are stable, and shows that accurate yet simple and easily understood relationships can be established by expressing the velocities at the surface in a Taylor series about a boundary grid point. This approach is successfully used in both the analysis (direct) and inverse cases (design).

When the present program is used in the design mode, the shape of the nose region (typically 6-10% of the chord) is specified and the pressure is prescribed over the remainder of the airfoil. This procedure is used for several reasons. First, the nose region must be accurately known in order to correctly fabricate an airfoil. Thus, by prescribing the nose shape, a possible major source of error is eliminated from the design procedure. Secondly, the boundary condition in the inverse region is ϕ_x and a starting value must be known. With the present scheme, this value is determined by the direct solution in the nose region and need not be estimated or iterated for. Third, in some cases the designer may wish only to modify the aft portion of the airfoil. This can be done with the present program since the switch point from direct to inverse can be set anywhere from about 6% chord to the trailing edge by an input variable. Finally, and perhaps most importantly, specification of the nose shape gives the designer a physical entitly wherby he can control the degree of closure at the tail. This will be shown later.

Any new numerical technique can only be verified by comparing its results with those previously obtained by other investigators. As suggested by Lock,[(6)] the NACA 0012 airfoil is an excellent test case because its shape can be prescribed analytically. Figure 2 compares analysis results obtained by the present method with those of Sells [(7)] for a lifting subcritical case. The two sets of data are always within two percent of each other and the lift coefficients agree exactly. In particular, notice the excellent agreement on the magnitude and location of the upper surface pressure peak.

For supercritical cases, comparison and verification is somewhat more difficult. However, a comparison with results obtained using Jameson's conformal mapping program with 192 points on the airfoil is shown on Fig. 3.

The present method results were obtained with medium grid which yielded 66 points on the airfoil surface. Notice that the lift and moment coefficients essentially agree exactly and that the pressure coefficients and shock location agree quite well. Similar verification of the accuracy of the present Cartesian grid program has been obtained for biconvex and NACA 63A006 airfoils. Based upon these results it is believed that the present approach is valid and quite accurate.

In order to verify the accuracy of the design mode of the program, the Cp distribution shown on Fig. 3 was used as input. The resultant slopes for the designed airfoil are shown on Fig. 4 and compared with the actual NACA 0012 slopes. The agreement is excellent even though there is a strong shock on the upper surface. For this case the computed surface ordinates were everywhere within 0.33% $(T/C)_{max}$ of the actual NACA 0012 ordinates. Thus, it is believed that the present design scheme is accurate and self-consistent.

As indicated above, in the present program the nose shape can be used by the designer to control tail closure. The procedure is demonstrated on Fig. 5, which shows three airfoils all designed for the same Cp distribution from 7% chord to the trailing edge. Airfoil No. 4 has an NACA 0012 nose shape but too thick of a tail. (The surfaces shown are displacement surfaces. The actual airfoil would be obtained by subtracting the displacement thickness from those ordinates.) Thus, a nonsymmetrical nose shape having a smaller leading-edge radius was used on Airfoil No. 5, which resulted in a much better tail size. Finally, for Airfoil No. 6 the lower surface nose region ordinates were raised by 0.001, and this led to an even thinner trailing edge.

Figure 6 shows another case in which the nose shape was used to control the trailing edge characteristics. The pressure distribution on this airfoil, which is the solid curves on Figure 7, was selected to have the same basic lift coefficient and lower surface pressure distribution as an NACA 0012 but without the strong upper surface shock wave. In this case, the nose shape is that associated with NACA 00XX airfoils; and, as can be seen, as the leading edge radius is increased, the tail opens up. Obviously any desired thickness of the trailing edge displacement surfaces can be achieved by adjusting the nose shape. Notice, also, that this adjustment does not require changing the desired inverse Cp distribution.

Now a severe test for a design program is whether or not an analysis or direct solution of the designed airfoil returns the design or inverse Cp distribution. Figure 7 compares the inverse Cp used to design airfoil No. 115 with that obtained from a direct solution (airfoil given) using the ordinates for No. 115. The excellent agreement tends to verify the validity and accuracy of the airfoils designed by the present program.

A final case is shown on Figure 8. An arbitrary pressure distribution, dashed line, having an upper surface Mach number plateau around 1.2 followed by a large jump at 76% chord was selected for the inverse input. On the lower surface, the Cp was chosen to maintain subsonic flow. However, a bucket selected according to the Stratford criteria was included to enhance lift. As indicated, the design program uses a backward difference scheme with the Cp input. Thus, in regions of large gradients the output, which is computed by a central scheme and should be more accurate, will be different.

Now in the course of the inverse solution, the trailing edge displacement surfaces that satisfied the input Cp were not parallel, and, thus, the inviscid solution required a rear stagnation point behavior. The actual inverse Cp (central differences), indicated by the solid line, shows this behavior. In addition, the upper surface discontinuity was smoothed. Examination shows a smooth supersonic bubble and indicates that upper surface decelleration, while rapid, is not due to a shock wave of any significant strength. Also shown is the result of a direct solution, which agrees well with the actual inverse Cp. The airfoil shown is the actual shape after the boundary displacement thickness has been subtracted. It is believed this result demonstrates that the present program can handle an "arbitrary" Cp input and will yield verifiable results consistent with physical reality, even if the input Cp does not.

In conclusion, it is believed that it has been shown that:

(1) It is not necessary to match the computational grid to the airfoil surface and that very accurate results can be obtained with a Cartesian grid. This may be important for 3-D calculations where mapping to the wing-body surfaces is frequently impractical.

(2) A design method having simultaneous airfoil update, a logical method for controlling trailing edge closure, and results that are consistent and physically correct has been created and incorporated into a working program.

References

1. Bauer, F., Garabedian, P., and Korn, D., Supercritical Wing Sections, Springer-Verlag, New York, 1972.

2. Steger, J. and Klineberg, J., "A Finite-Difference Method for Transonic Airfoil Design," AIAA Journal, Vol. 11, No. 5, May 1973, p. 628-635.

3. Tranen, T. L., "A Rapid Computer Aided Transonic Airfoil Design Method," AIAA Paper 74-501, June 1974.

4. South, J. and Jameson, A., "Relaxation Solutions for Inviscid Axisymmetric Transonic Flow over Blunt and Pointed Bodies," Proc. AIAA Computational Fluid Dynamics Conference, Palm Springs, Calif., July 1973, pp. 8-17.

5. Jameson, A., "Iterative Solution of Transonic Flows over Airfoils and Wings, Including Flows at Mach 1," Comm. on Pure Appl. Math., Vol. 27, pp. 283-309, 1974.

6. Lock, R. C., "Test Cases for Numerical Methods in Two-Dimensional Transonic Flows," AGARD Rept. R-575-70, 1970.

7. Newmann, P. A., Theoretical Aerodynamics Branch, NASA Langley, private communication, October 1974.

Acknowledgment

This research was partially supported by NASA Grant NGR-44-001-157, the NASA-ASEE Summer Faculty Fellowship Program, and Texas Engineering Experiment Station.

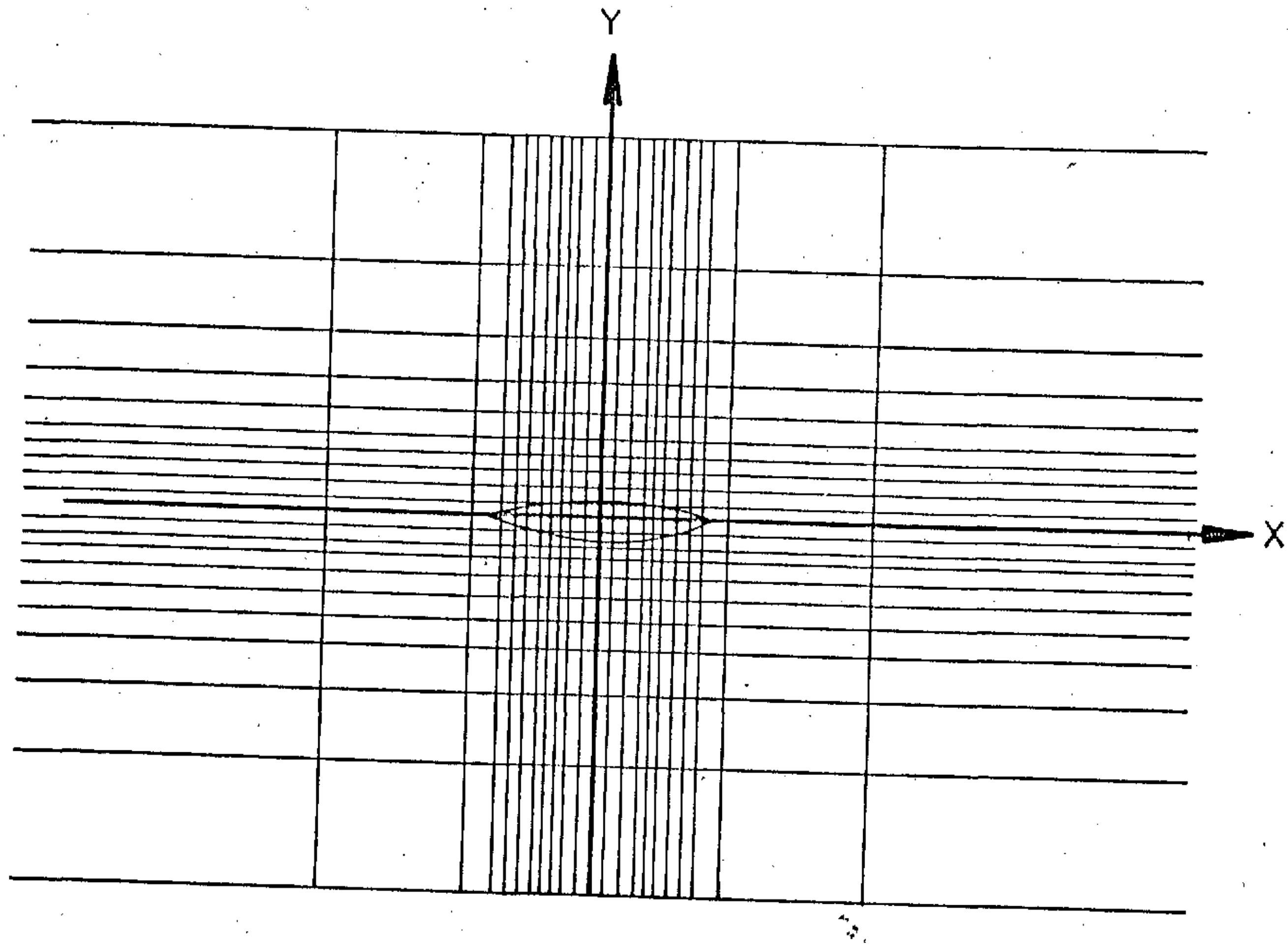


Figure 1 -- Typical Grid System (Schematic)

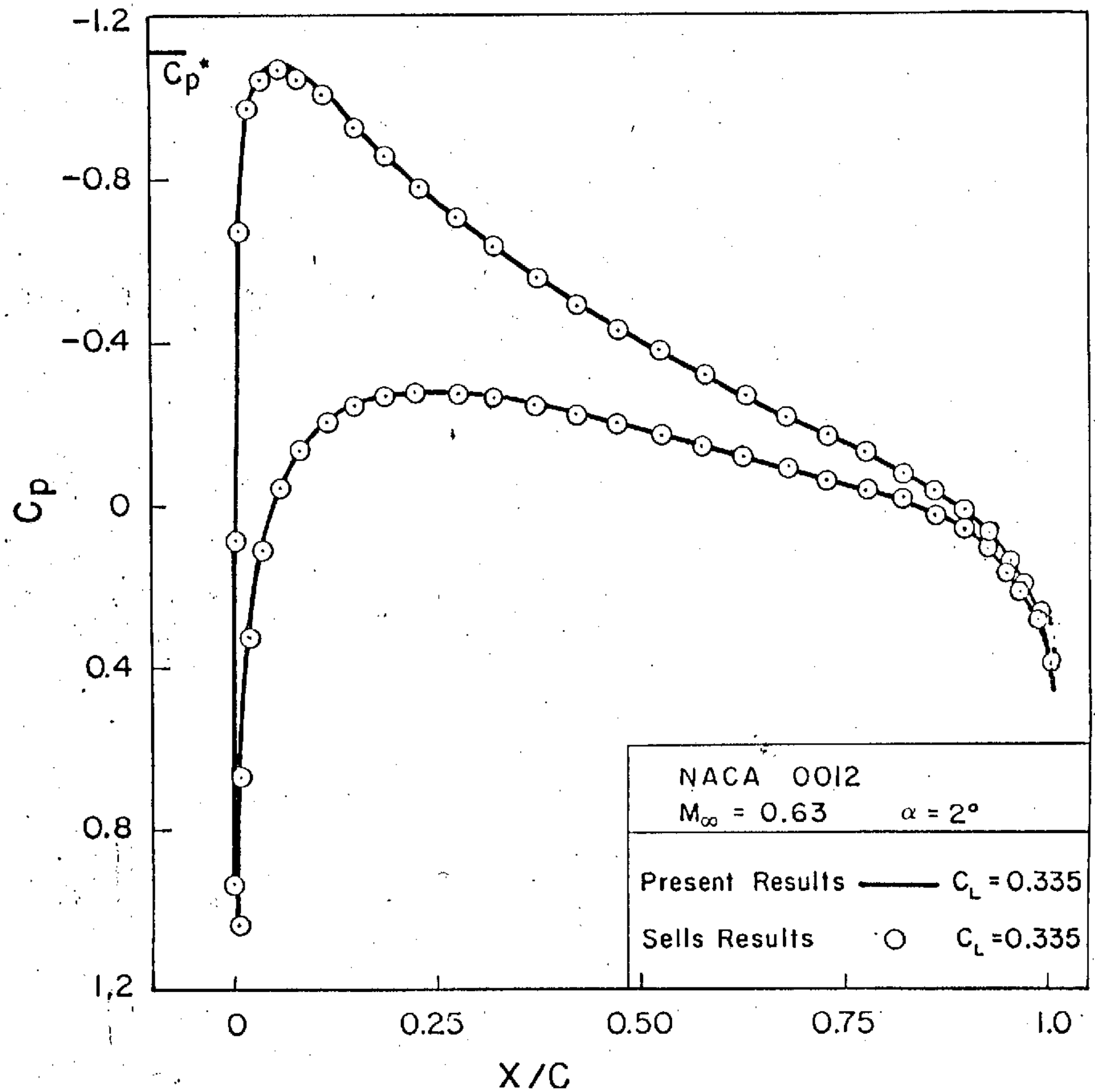


Figure 2 -- Comparison with Sells for Subcritical Lifting Case

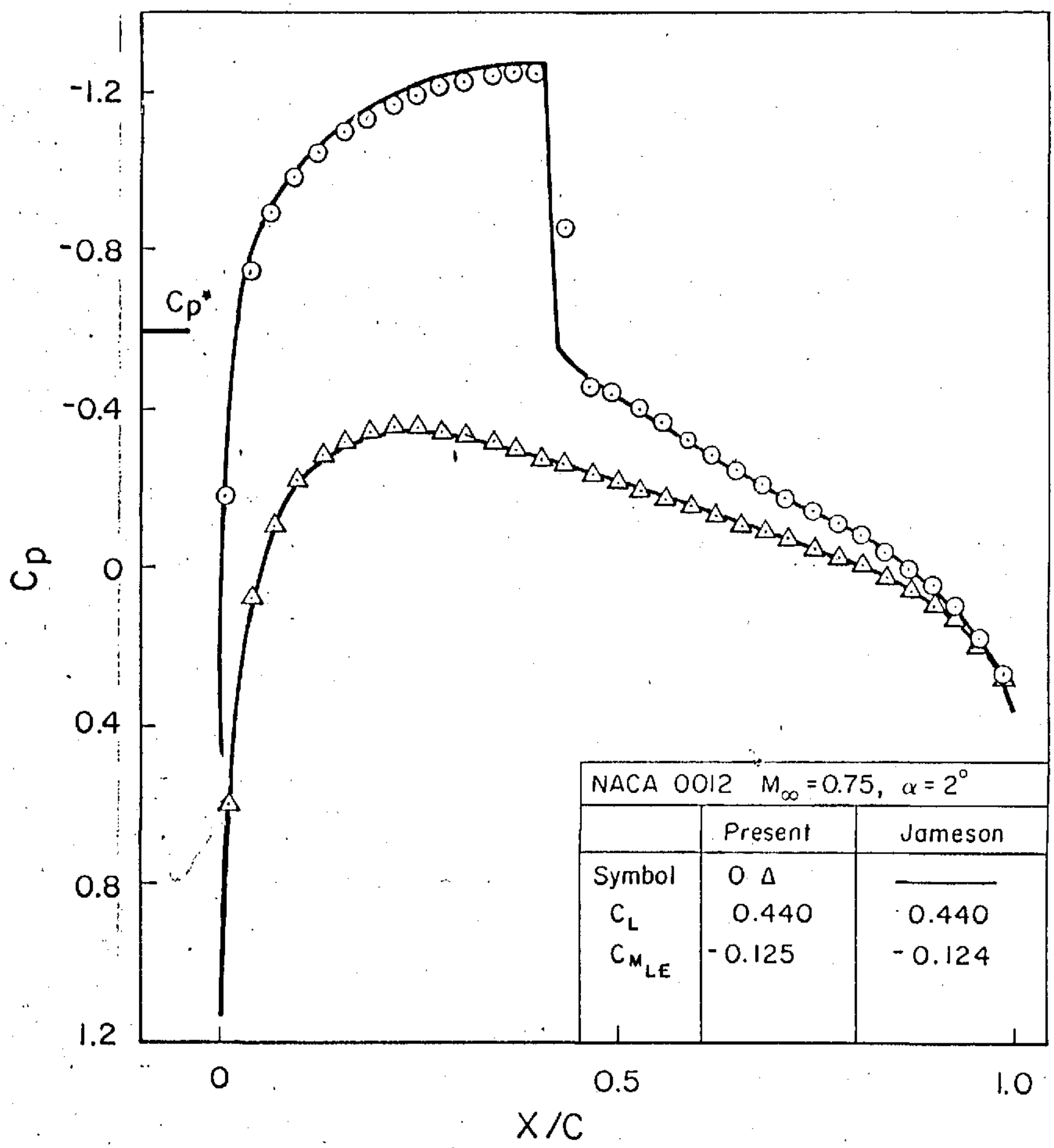


Figure 3 -- Comparison with Jameson for Supercritical Lifting Case

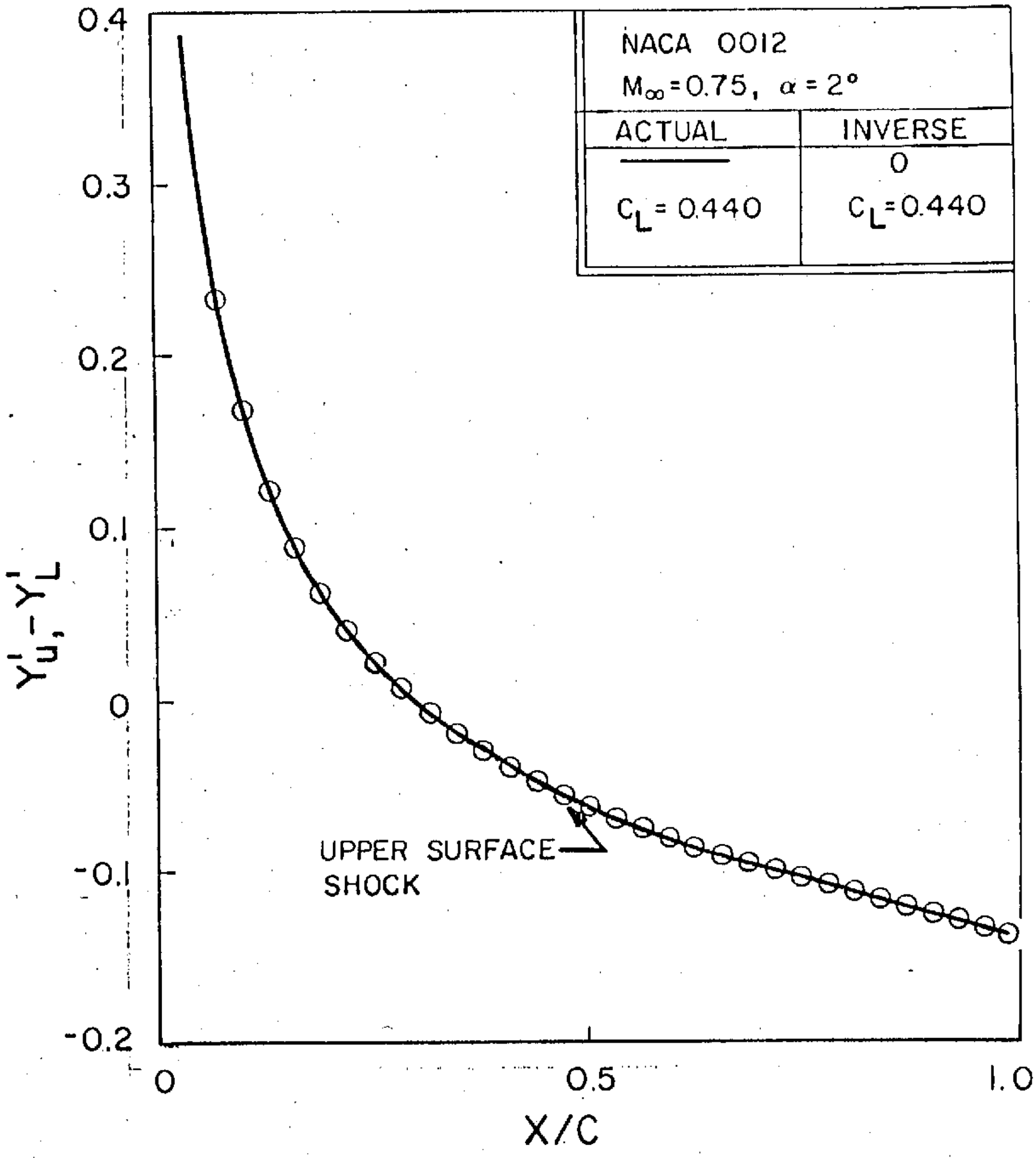


Figure 4 -- Comparison of Actual Airfoil Slopes with Those Computed by Design Program

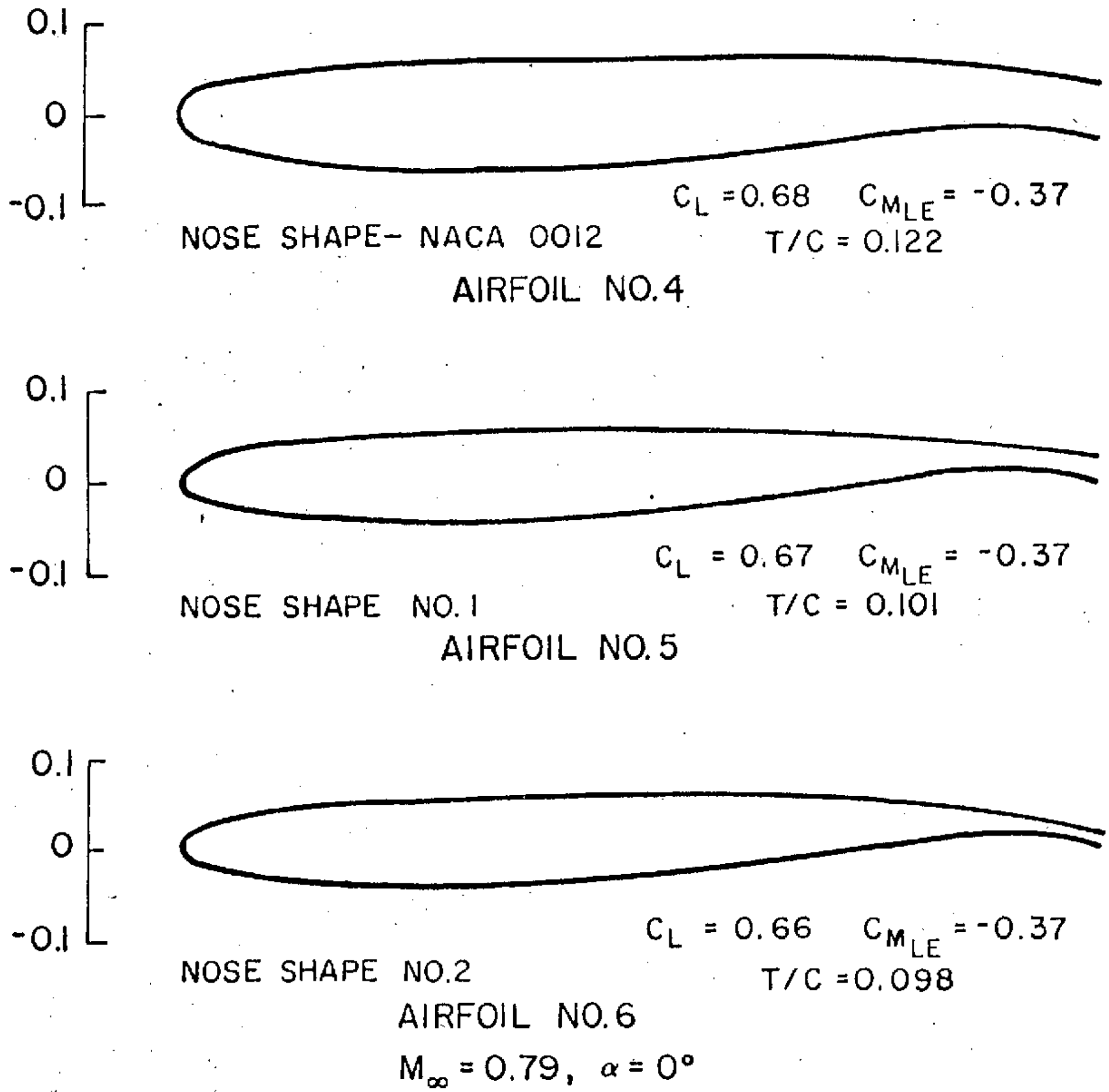
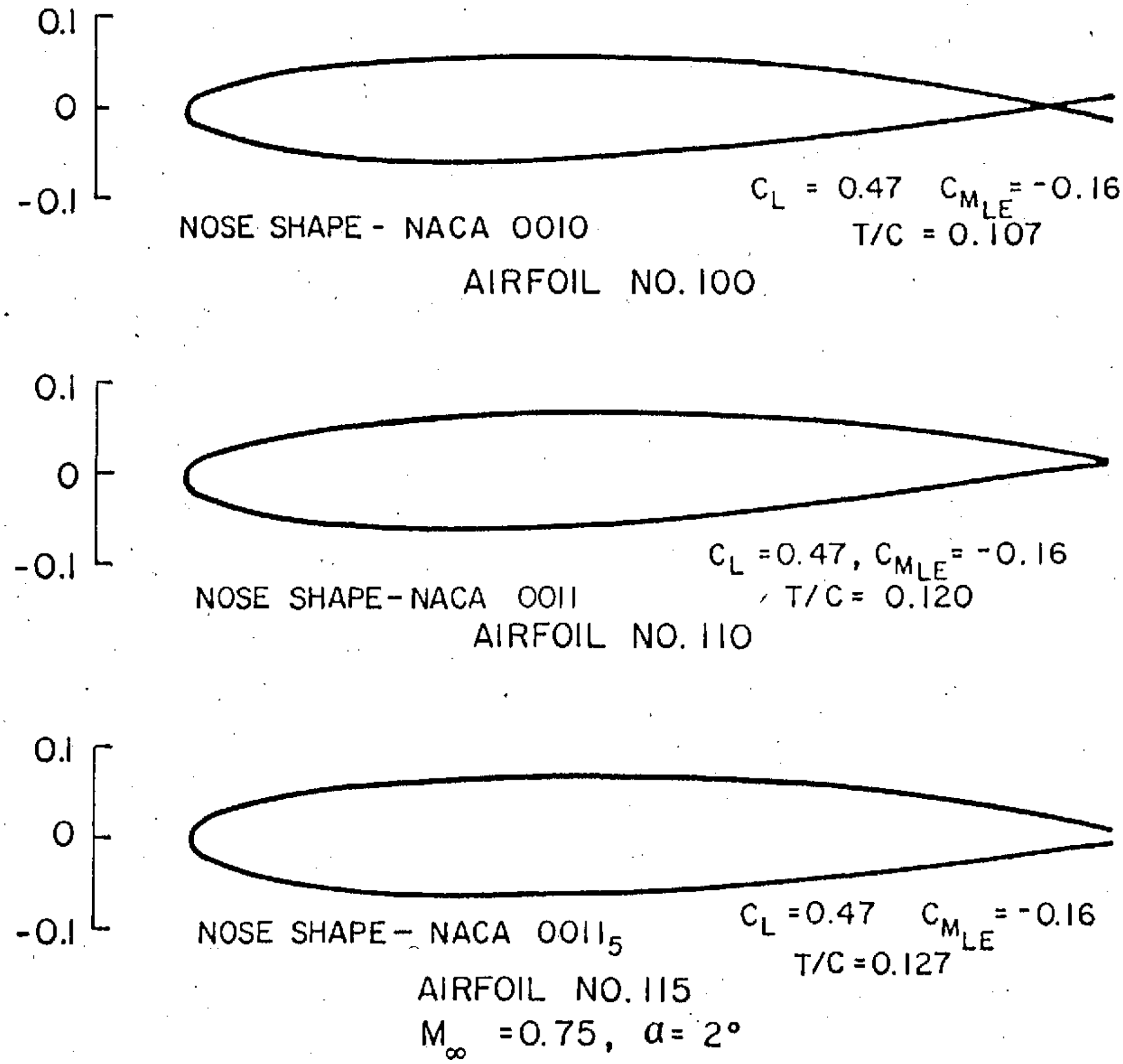


Figure 5 -- The Use of Nose Shape to Control Trailing Edge Closure, Example 1



<u>Figure 6</u> -- <u>The Use of Nose Shape to Control</u>

<u>Trailing Edge Closure, Example 2</u>

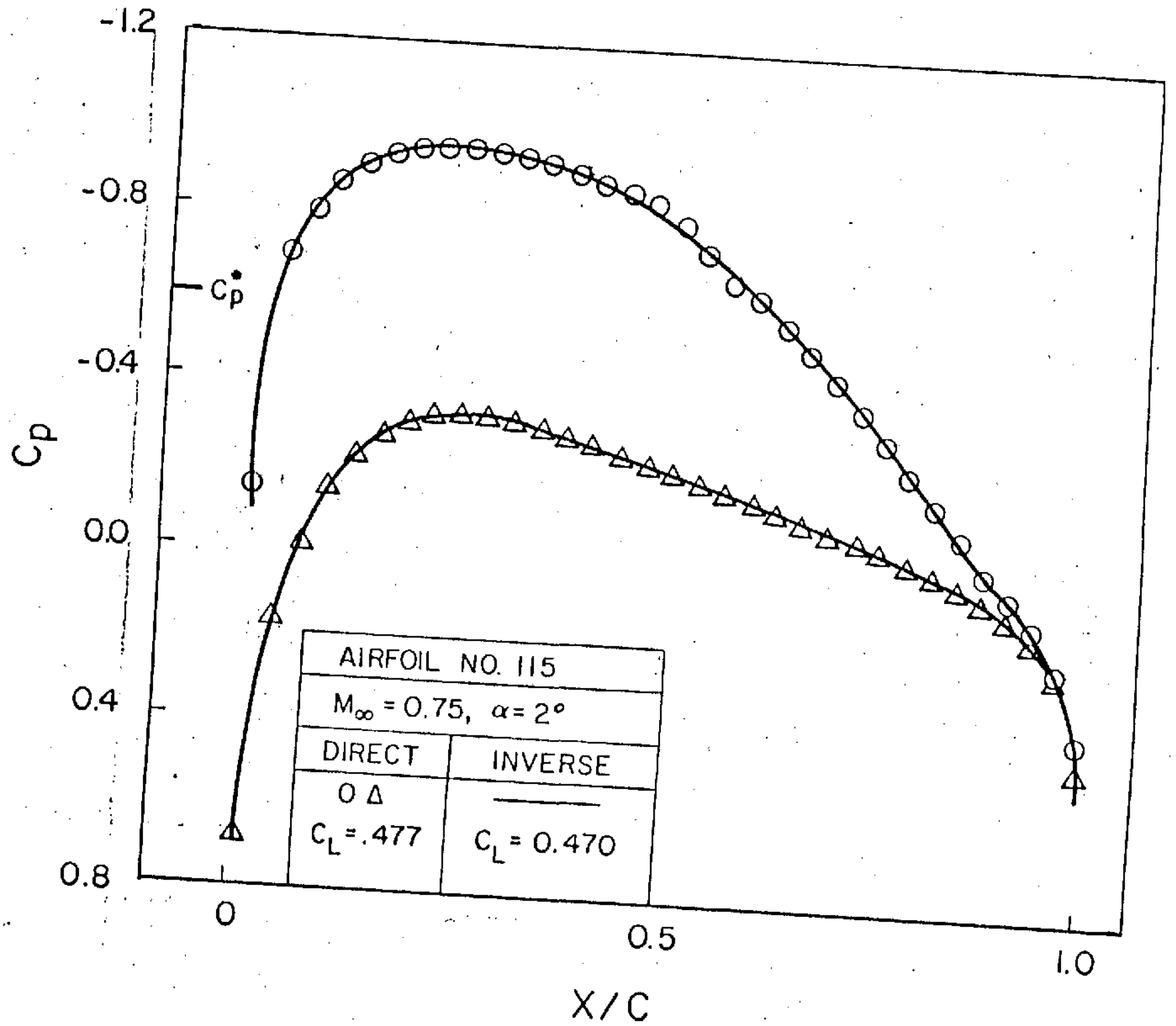


Figure 7 -- Comparison of Inverse C_p Distribution with that Obtained by Analysis of Designed Airfoil

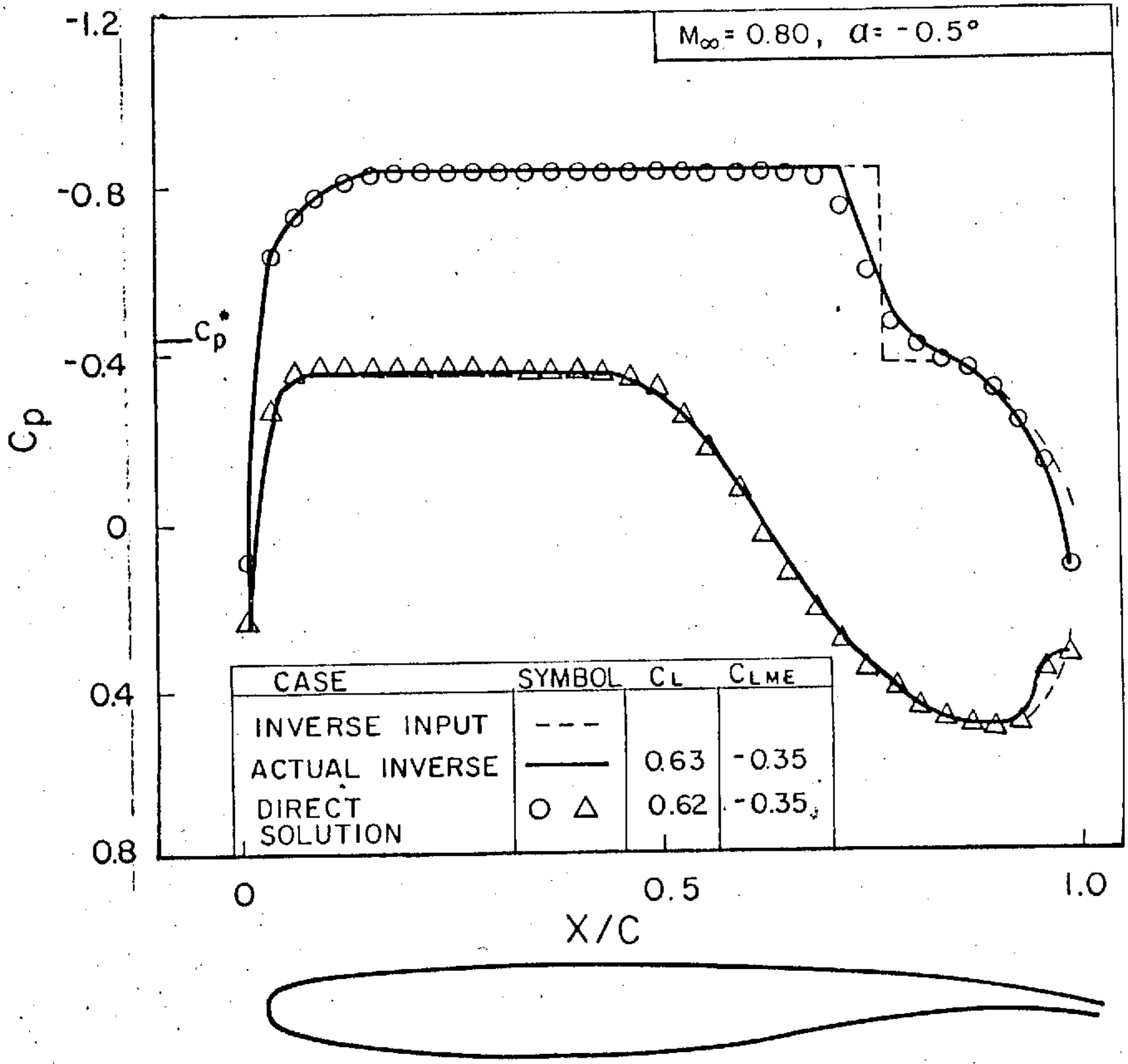


Figure 8 -- Comparison of C_p Distributions