INVERSE TRANSONIC AIRFOIL DESIGN METHODS
INCLUDING BOUNDARY LAYER AND VISCOUS
INTERACTION EFFECTS

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I. Introduction

During the past six months, considerable progress has been made under the present grant. Since the primary purpose of the grant is to include viscous interaction effects in the transonic airfoil design and analysis program[1,2] developed under grant NGR-44-001-157, various boundary layer computational schemes have been investigated as to their suitability and accuracy. As a consequence, the Nash-McDonald[3] method was selected, and it has been appropriately modified and included in both the analysis and design modes of the program. The results to date are promising, and a brief discussion of these efforts and results is presented in the following sections.

II. Discussion of Research

a. Boundary Layer Scheme

Several different boundary layer methods have been examined in order to determine those methods most suitable for inclusion in the present computer program. Since the boundary layer has to be computed approximately every ten relaxation cycles, finite difference methods were eliminated as being too time consuming and costly, and emphasis was placed on integral methods. Primarily as a result of the Stanford Conference[4] and the success of Bauer et.al.[5], three methods were selected for detailed study--the Walz Method II[6], the Nash-McDonald Method[3], and Nash-McDonald with smoothing[5].

The Walz method can handle compressible laminar and turbulent boundary layers, permits wall temperature variations, and allows for the inclusion of a transition criteria. It yields accurate results but uses a variable step size that is determined by the solution. In addition, it numerically fails at separation due to the nature of some of the empirical equations it utilizes. The Nash-McDonald approach, on the other hand, is not computationally limited by separation and is approximately six times faster. However, its results sometimes exhibit oscillations, which fortunately, can be eliminated by smoothing.

Figure 1 compares displacement thickness predictions from each of these methods with those obtained by Bavitz[7] using the method of Bradshaw[8]. These results are for a Garabedian-Korn 75-06-12 airfoil at Mach 0.702, 1.10° angle of attack, and a Reynolds number of 21.18×10^{6}. Notice that even on the exaggerated scale of Figure 1, the predictions are essentially identical. Further, all four methods predict separation at essentially the same location. Based upon these and other results, upon speed, and upon the ability to predict boundary layer properties at *a priori* selected coordinates, the Nash-McDonald method with smoothing was selected for incorporation into the present transonic airfoil program.

b. Design Program

In the design mode, the actual airfoil shape computed by the program is the displacement surface, and the actual airfoil is determined by subtracting the displacement thickness calculated by the boundary layer scheme from this surface. If, however, the input pressure distribution does not lead to a displacement surface having a cusped trailing edge, the inviscid program will yield a final pressure distribution having a rear stagnation point. Physically, such a distribution would lead to separation and would induce near the trailing edge a large displacement thickness. While this effect is reasonably correct on conventional airfoils, on supercritical airfoils having lower surface pressure buckets trailing edge separation would relieve the adverse pressure gradient, lower the displacement thickness, and lead to a trailing edge pressure far from stagnation.

In order to account for this phenomena, a simple trailing edge modification has been introduced into the viscous calculation package of the design program. If upper surface separation occurs prior to last computational point before the trailing edge, the lower surface pressures at the three points before this point are used to determine a curve fit and to recompute the trailing edge pressure. This curve fit is also used to recompute the last lower surface pressure value. On the upper surface, the pressure distribution is modified by assuming a linear variation from the separation point pressure to the new base pressure. This modification is shown on Fig. 1(a). Then, the boundary layer is recomputed with the modified distribution and the resultant displacement thickness is used to determine the ordinates of the actual airfoil. Notice that this modification would not extensively change the pressures on a conventional airfoil. Of course, further tests will be required in order to determine if this approach is reasonable.

c. Viscous Interaction in the Analysis Program

The Nash-McDonald boundary layer analysis method with smoothing has been programmed and incorporated into the direct analysis part of the program. The approach utilized is essentially the same as Bauer et.al.[(5)] except that the boundary layer is computed every ten relaxation cycles and that the computational points coincide with the inviscid x-coordinates in all grids. This latter approach seems to assist the convergence between the inviscid and viscous calculations.

In order to test the present scheme comparisons have been made with the detailed results presented by Bavitz[7], and these comparisons are shown on Figures 2-4. The results shown were obtained on a medium grid (49x49) that yields 66 pressure points on the airfoil surface. In general, the agreement is reasonable, even though the medium grid did not resolve the sharp shock. Figures 2 and 4 are for different angles of attack because the present coordinates for the 75-06-12 airfoil were obtained from Ref. (5) while Bavitz based his coordinates on different chord line. Thus, the required angles of attack will be different. Nevertheless, good agreement on lift and moment coefficients is obtained; and the present drag coefficient prediction is close to those measured experimentally at these conditions[9] (i.e. near 0.0100).

Figure 3 compares the predictions for displacement thickness for the same conditions. Both results agree on the location of separation and are in reasonable agreement on the actual values. The Nash-McDonald result, however, does predict the expected decrease in the lower surface displacement thickness near the trailing edge, which is prohibited in the Bavitz scheme.

Comparisons between results obtained with the present viscous interaction approach and experimental data[9] obtained at the NAE for subcritical flow past a 75-06-12 airfoil are shown on Figure 5. In the tests, the lift and moment coefficients obtained by pressure integration and by force balance measurement differed slightly, yielding in the case of C_L 0.44 and 0.49, respectively. Also, the drag value was measured by a wake rake while the theoretical prediction was obtained by pressure integration. As can be seen, the agreement between the measured and theoretical aerodynamic coefficients is quite good, particularly when it is realized that the theoretical angle of attack had to be estimated.

A high lift supercritical case comparison is shown on Figure 6. Again, the airfoil is a Korn 75-06-12. On the figure, the experimental results[10] are represented by the symbols and the theory, obtained on a 49x49 grid, by the solid line. With the exception of the drag coefficient, all of the aerodynamic coefficient predictions are in reasonable agreement with the measurements. It is believed that even better agreement could be obtained for the pressure distributions by further adjustments in the theoretical freestream Mach number and angle of attack.

Now one of the features of the Korn airfoil is that near its design point (Mach No. = 0.75) its upper surface pressure distribution is characterized by multiple shocks. These shocks are difficult to detect theoretically and pose a severe test for any analysis technique. Figure 7 shows such a shock system obtained using a fine 97x49 grid, which yields 130 pressure points on the airfoil. In order to resolve these weak shocks with viscous interaction, 400 iterations were required on the fine grid. While this case does not correspond exactly to any NAE test, comparison with a close case[9] indicates that the aerodynamic predictions are quite good. (Although the predicted drag coefficient is high.) Also, the shock locations etc. do fit the pattern exhibited by NAE pressure distributions[10]. Figure 8 shows the corresponding boundary layer properties.

In examining cases near the design point, Bauer[5] et.al. discovered that occassionally the double shock system was not resolved on a medium grid and that many iterations on the fine grid were required to achieve convergence. A similar phenomena has been discovered with the present program, and it is illustrated on Figure 9. Here the medium grid converged in 212 relaxation cycles and had exhibited no essential changes for the last 100 cycles. On the other hand, like Bauer[5] et.al. the fine grid required 400 cycles. While Ref. (5) attributes this behavior to the artificial viscosity introduced by the difference scheme, this author believes that the inclusion of the boundary layer is just as instrumental. This belief is based on the fact that similar weak double shock systems have frequently been found with the medium grid for pure inviscid flows. Hence, more investigation will be required before the origin of this numerical phenomena is determined. Nevertheless, it should be noted that lift, drag, etc. are usually quite well predicted on a medium grid, and that frequently the results from a medium grid computation will suffice for engineering comparisons and studies.

As implied above, one of the primary uses of a viscous analysis program would be to estimate the aerodynamic coefficient characteristics of an airfoil. For example, the results of several computations near the design Mach number of a 75-06-12 airfoil are shown on Figures 10 and 11. These predictions, when compared to experiment, all show the correct trends, including the dip in the moment coefficient near a C_L of 0.7. The drag values are, however, too high, but they do show the correct behavior. These high values for C_D will be discussed later.

Finally, Figure 12 shows the beginning of a drag versus Mach Number plot for the 75-06-12 airfoil. While, once again the C_D values are high, the trend of these results is in agreement with experiment[9].

Based upon the above results, it is believed that the only major problem so far with the present scheme is the prediction of too high values for C_D, and even there the trends are correct. Nevertheless, an attempt will be made to correct these values. In the present scheme, the skin friction drag is determined by the Squire-Young correlation and the wave drag is computed via integration of the pressure. Each of these may be in error, and in the future the Squire-Young formula will be checked by actually integrating the skin friction. Likewise, the method of integrating the pressure will be improved, particularly in the leading edge region. In this manner, it is believed that the drag predictions will be improved.

III. Future Work

In the next reporting period, the following topics, among others, will be investigated:

1. A suitable trailing edge correction will be developed and incorporated into the analysis program in order to handle those cases having large trailing edge separation.
2. Green's lag-entrainment method of boundary layer analysis will be investigated and possibly adopted for use in the program. In this case primary emphasis will be on the design case.
3. An attempt will be made to determine more explicitly why weak shocks are not found on the medium grid.
4. The drag computational scheme will be improved so that accurate values can be found.
5. The development of suitable documentation and manuals for the computer programs will be initiated.

IV. Publications

The following has been partially supported by this grant:

"Transonic Airfoil Analysis and Design Using Cartesian Coordinates," Journal of Aircraft, (to be published).

V. References

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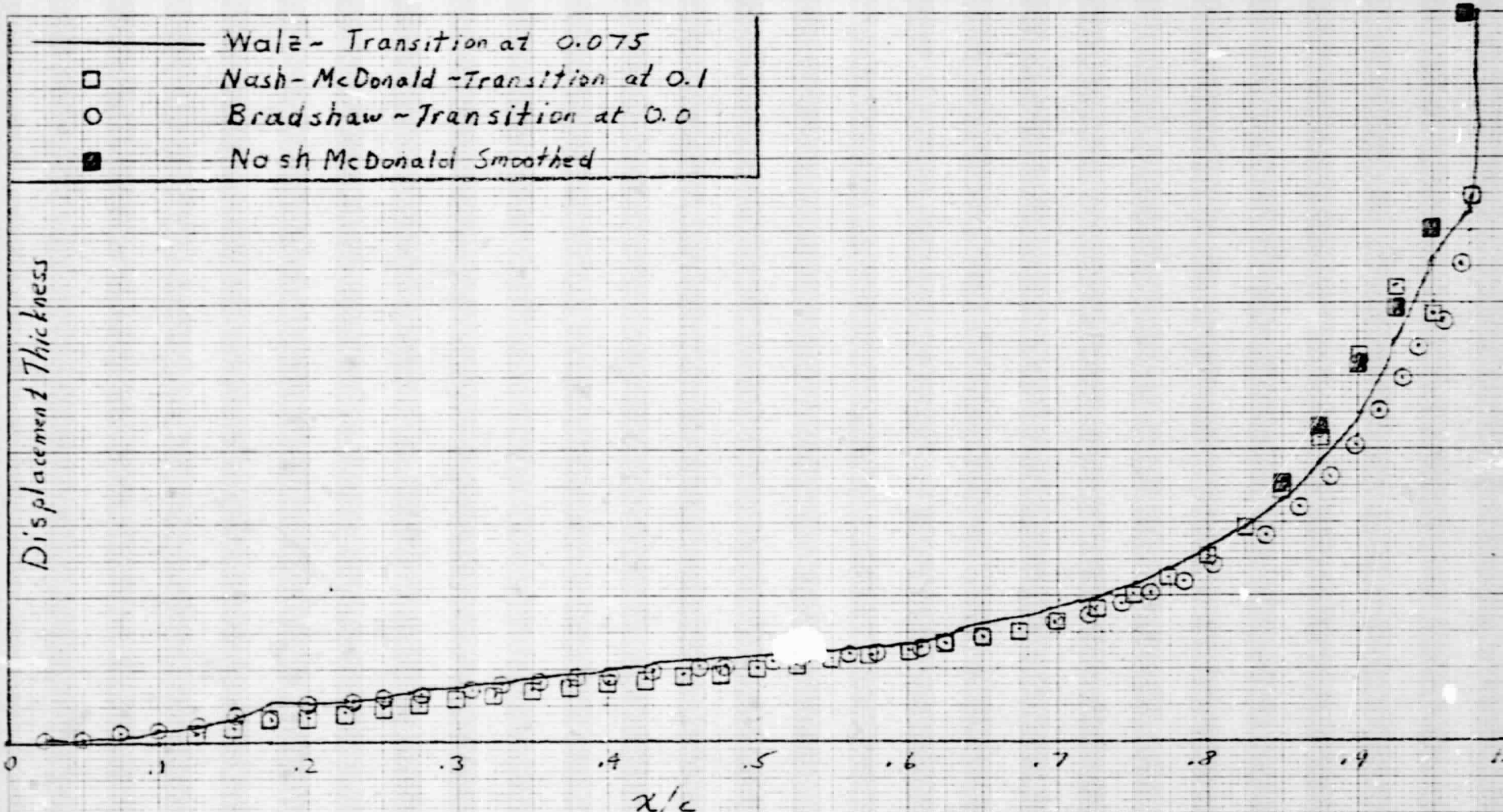
9. Kacprzynski, J.J., "An Experimental Analysis and Buffet Investigation of the Shockless Lifting Airfoil No. 1, NRC-NO-13673, Aero. Rept. LR-569, Aug. 1973.
10. Evans, W., MacKenzie, D., and Romanowski, R., "The Design, Test, and Evaluation of Shockless Transonic Airfoils," Final Report Contract N00019-72-C-0358, Sept. 1973.



Comparison of Boundary Layer Predictions

Figure 1

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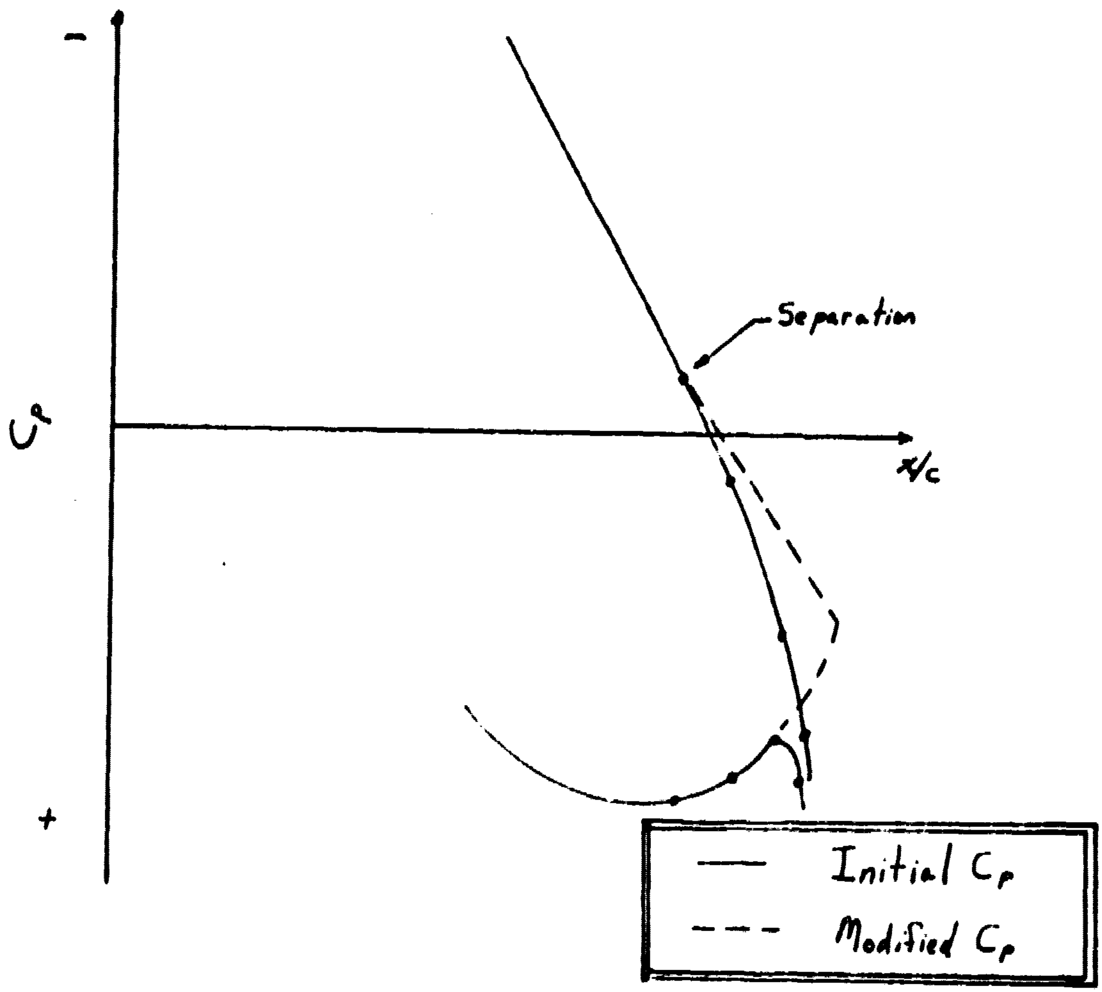
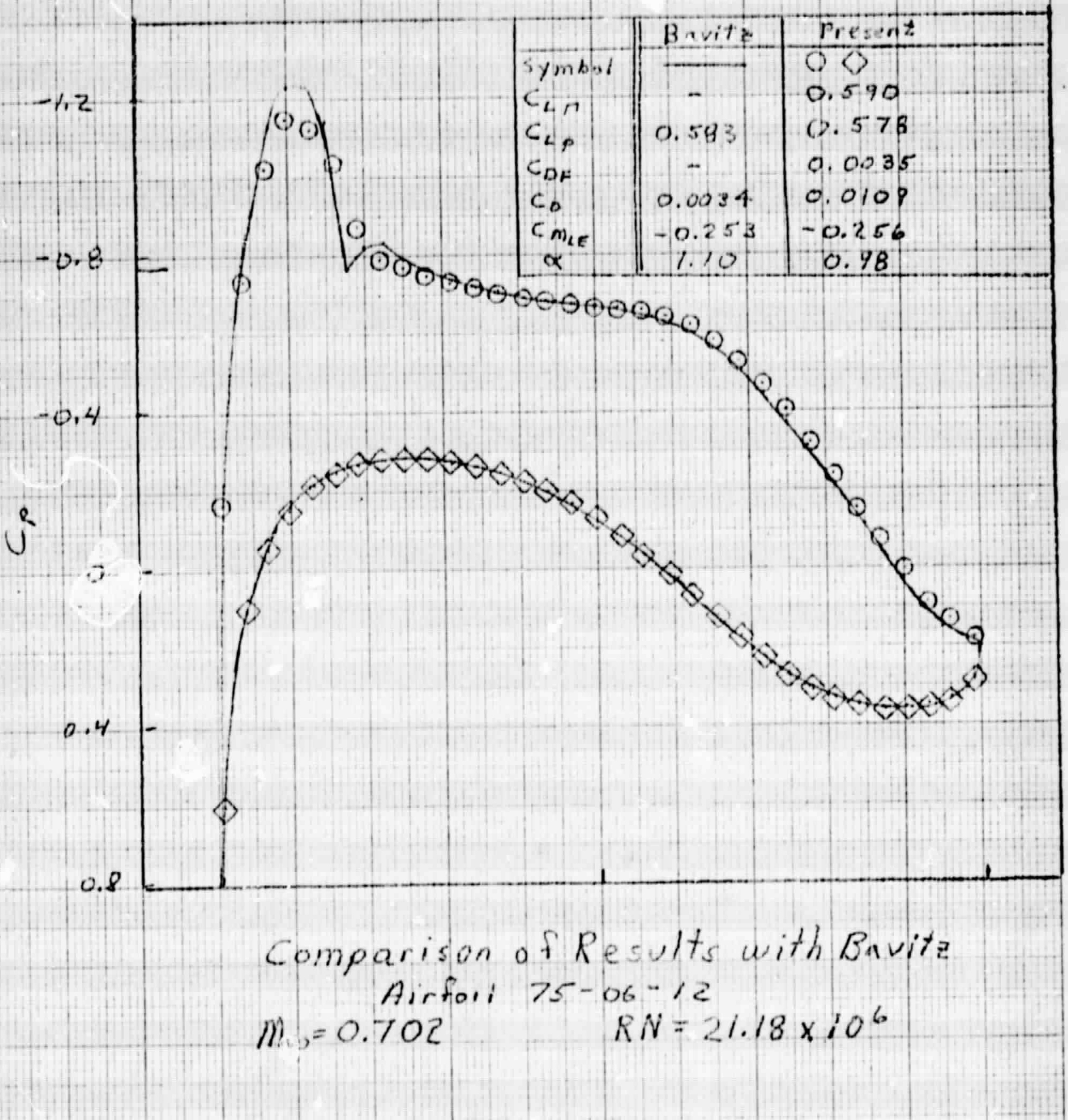


Trailing Edge Modification (Design Case)

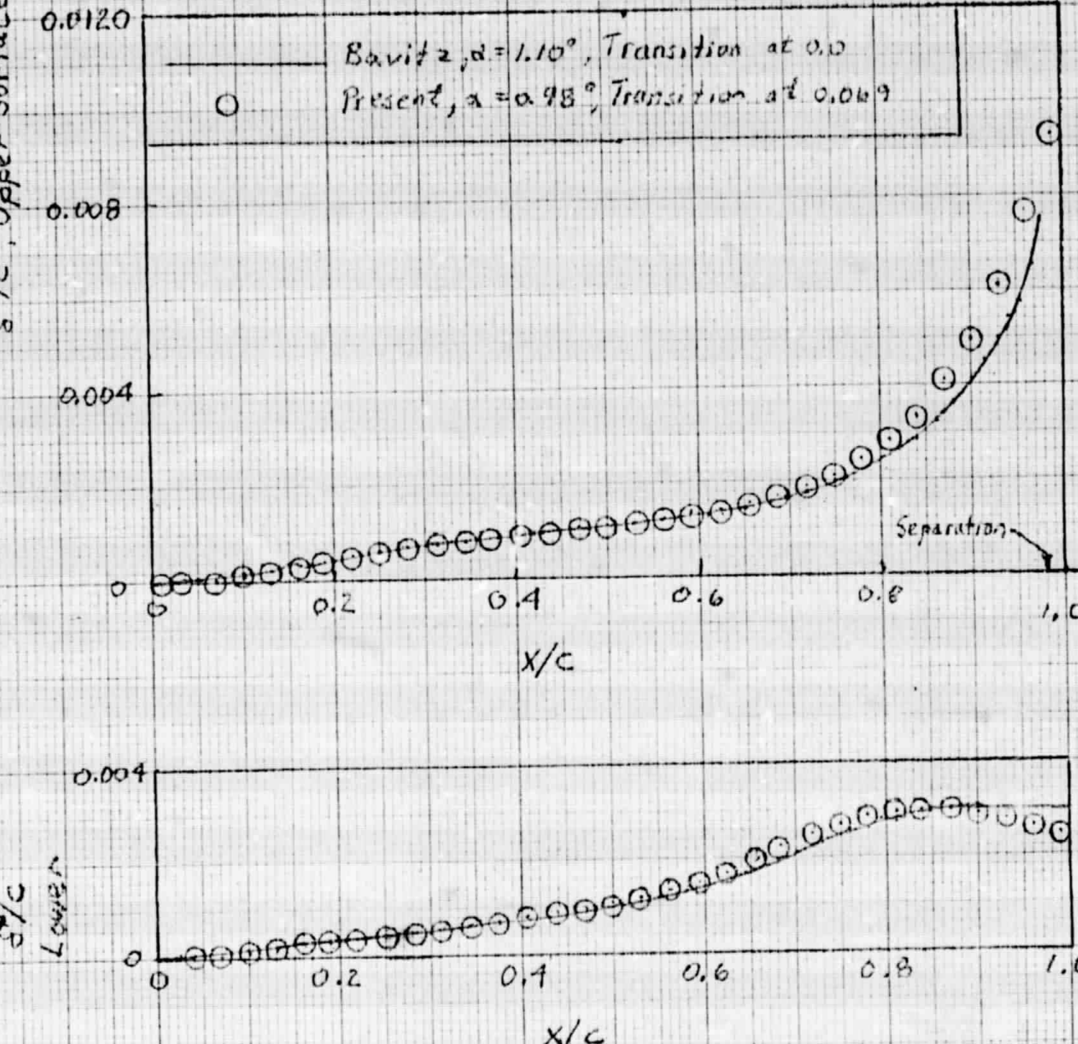
Fig. 1(a)

LRC



Comparison of Results with Bavitz
Airfoil 75-06-12
$M_\infty = 0.702$ RN = 21.18×10^6

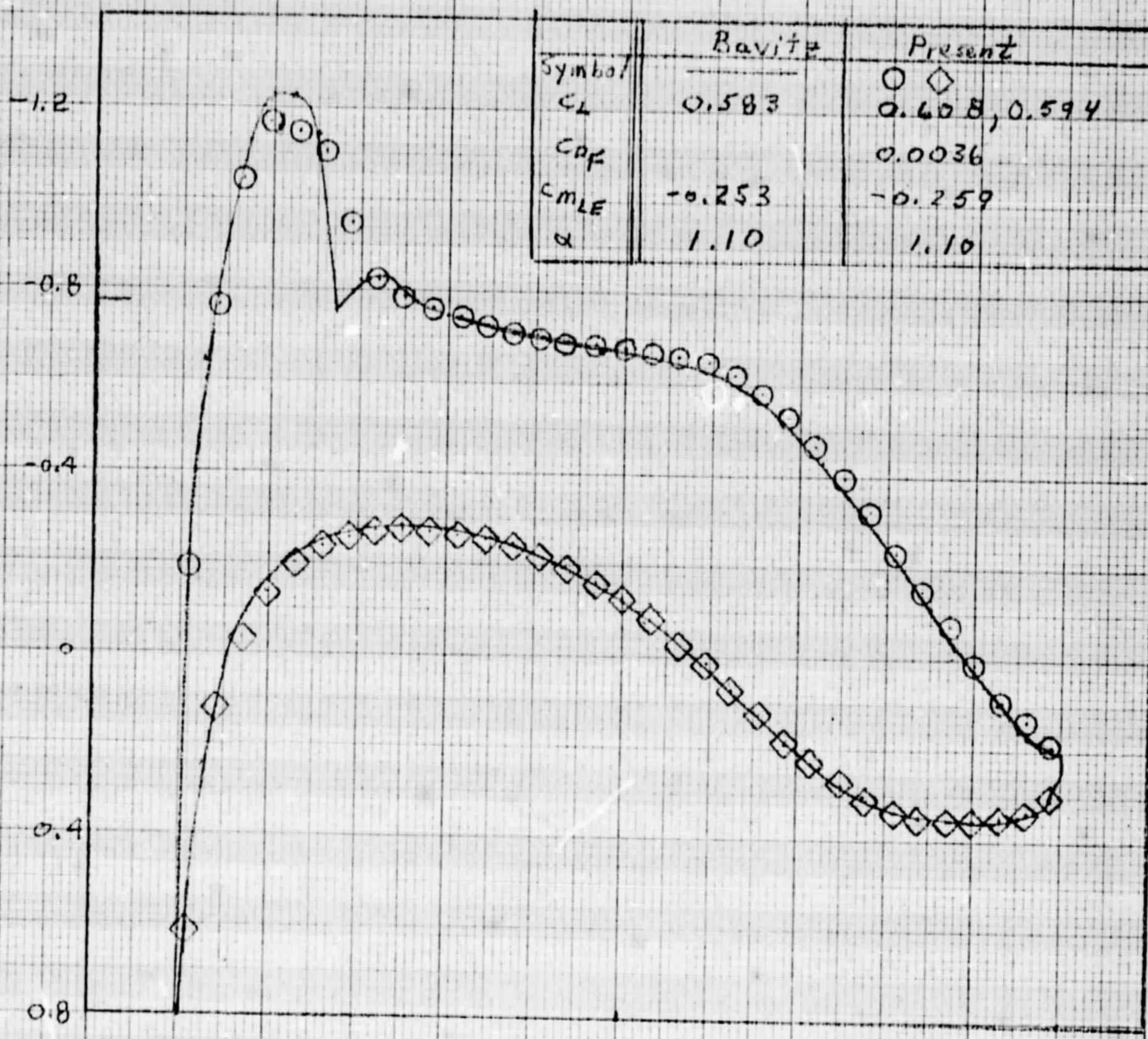
Figure 2



Comparison of Results with Bavitz
Airfoil 75-06-12
$M_\infty = 0.702$ $\alpha = 0.98°$ $RN = 21.18 \times 10^6$

Figure 3

JRC



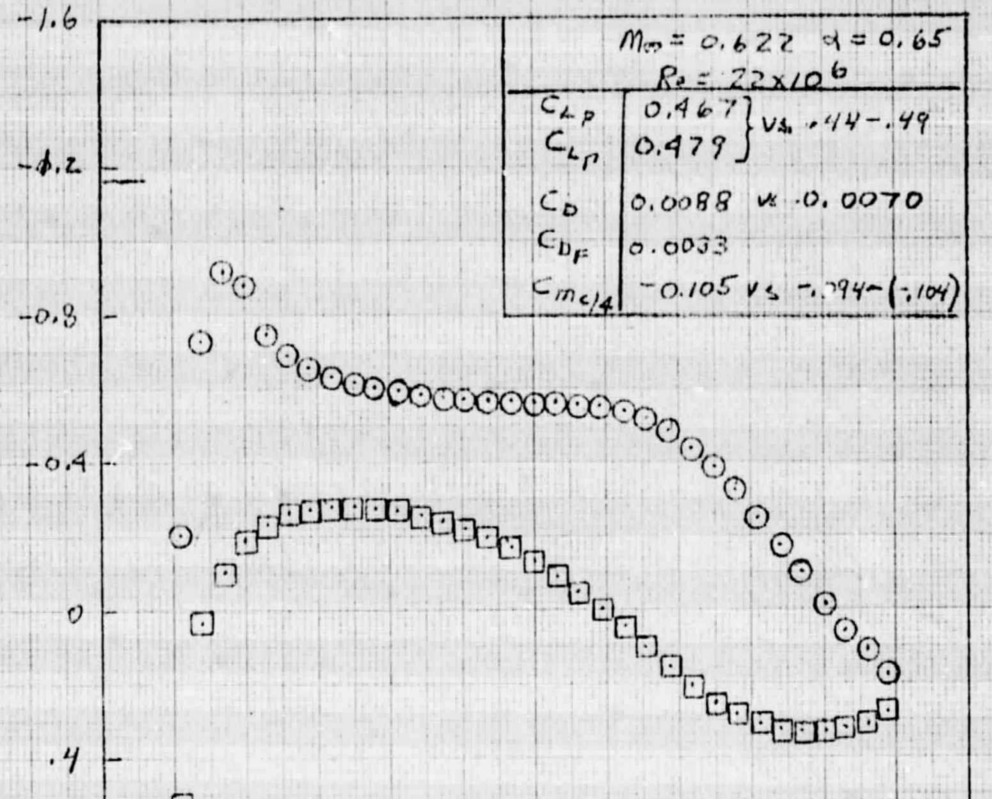
Comparison of Results with Bavitz
Airfoil 75-06-12

$M_\infty = 0.702$ $\quad$ $RN = 21.18 \times 10^6$

Figure 4

DDC

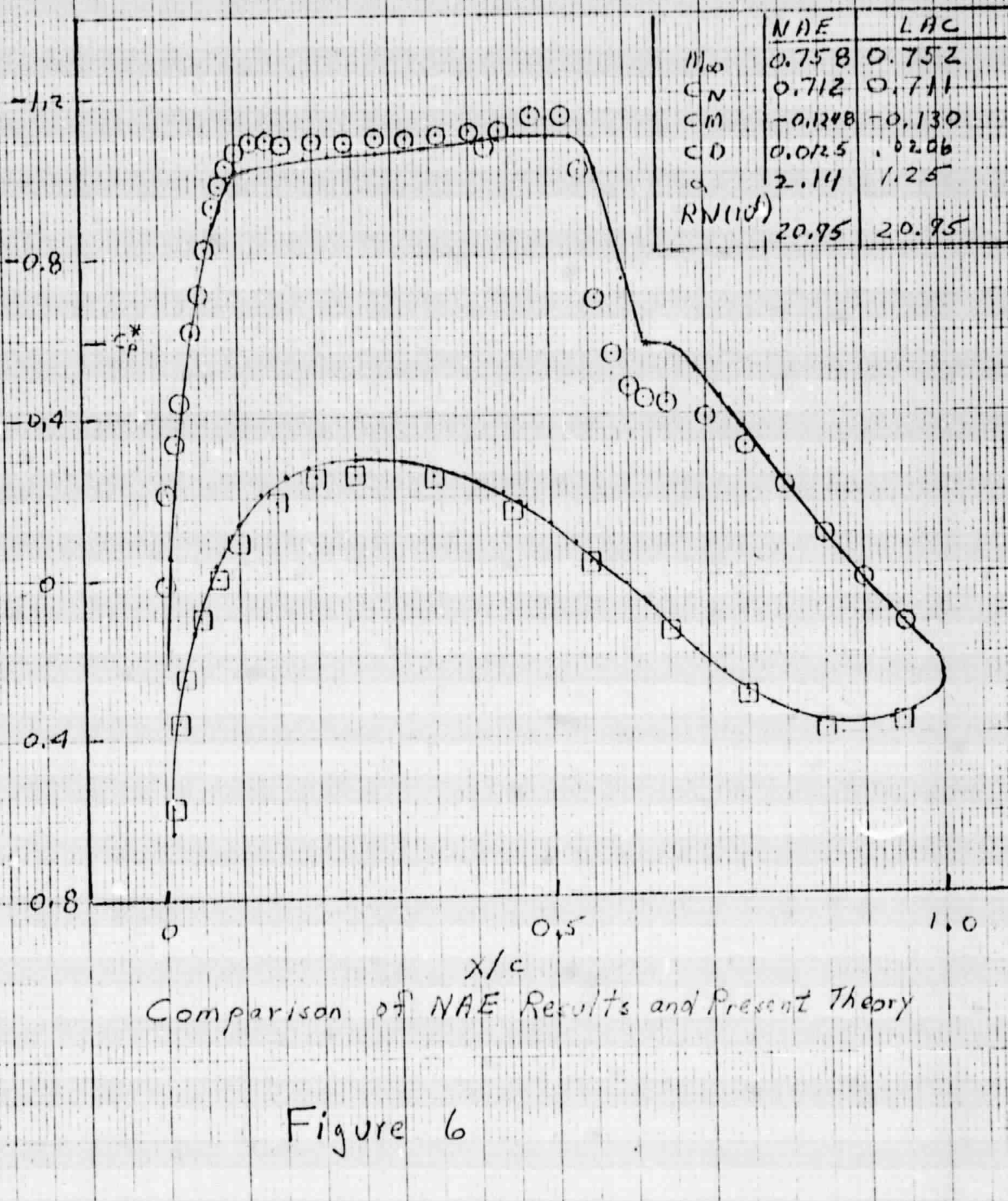
535



Comparison of NAE and Present Results

Figure 5

LAL



Comparison of NAE Results and Present Theory

Figure 6

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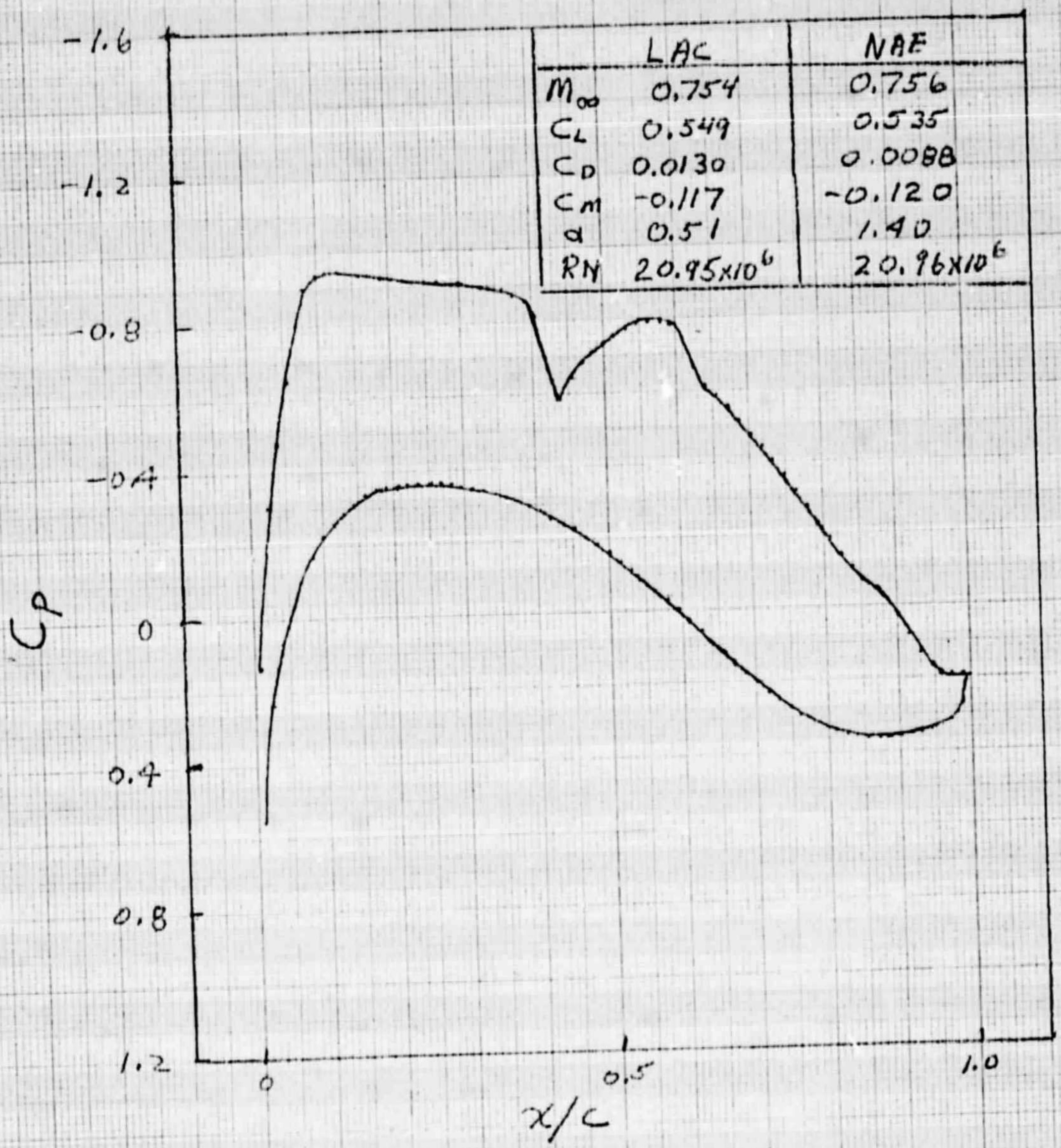

Fine Grid Results

Figure 7

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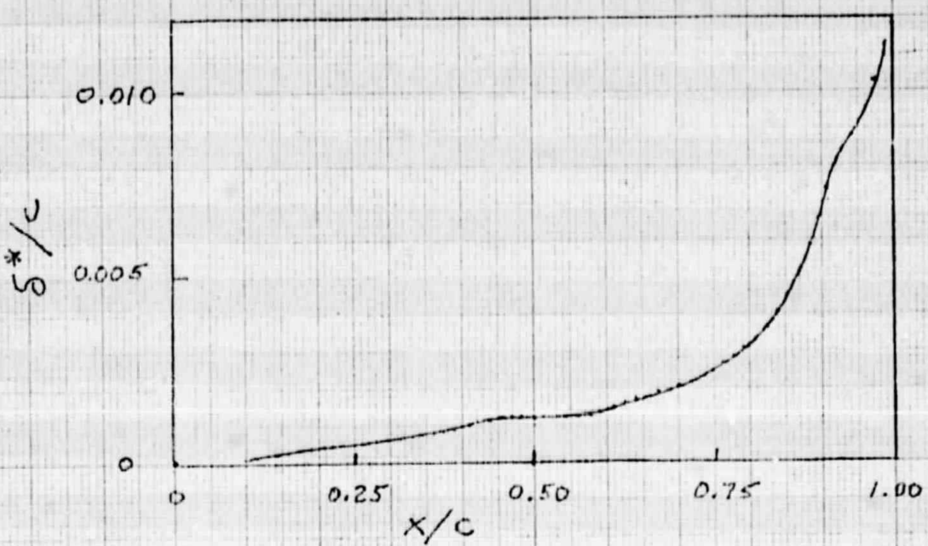
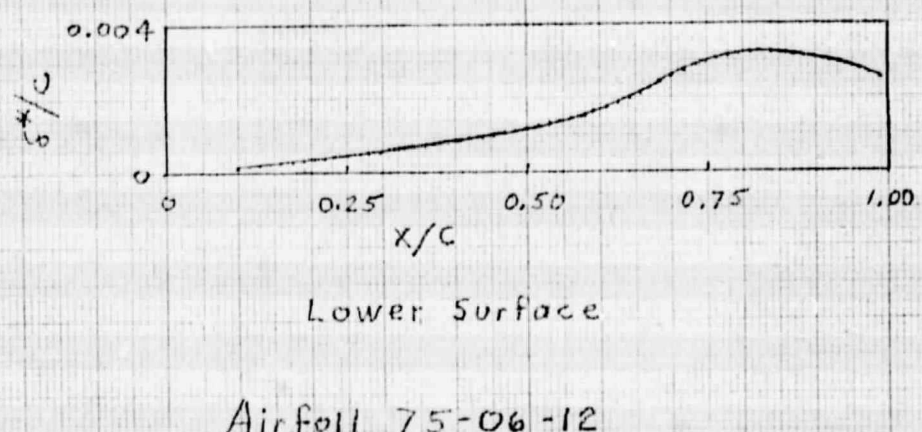


Upper Surface



Lower Surface

Airfoil 75-06-12

$M_\infty = 0.754$ CL= 0.549 , RN= 20.95×10^6

Figure 8

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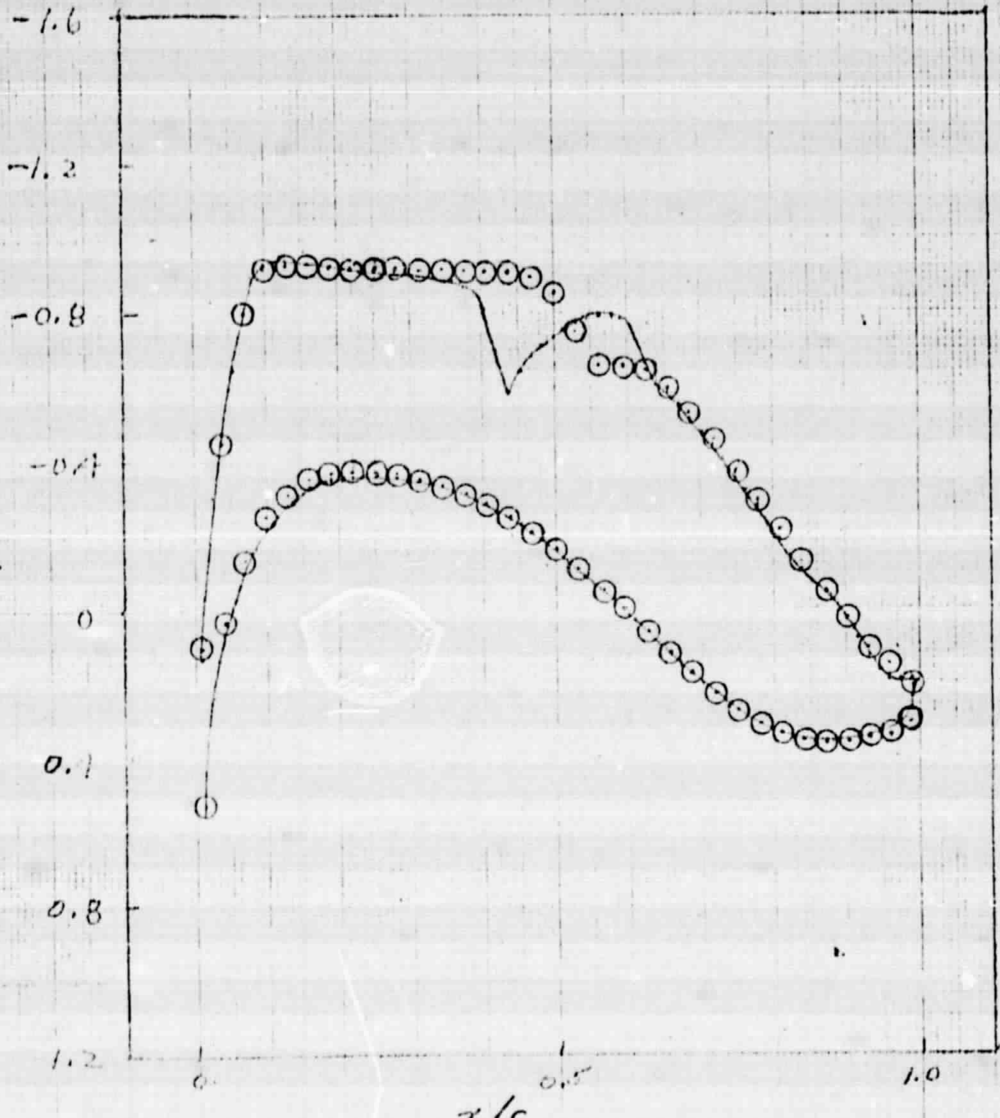

Airfoil 75-06-12 M=0.754 ALP=0.51 RN=21 Million

— Fine Grid MxN= 97x49 CL=.549 CD=.0130 CM= -0.117

O Medium Grid MxN= 49x25 CL=.568 CD=.0131 CM=-.124

The Effect of Grid Size

Figure 9

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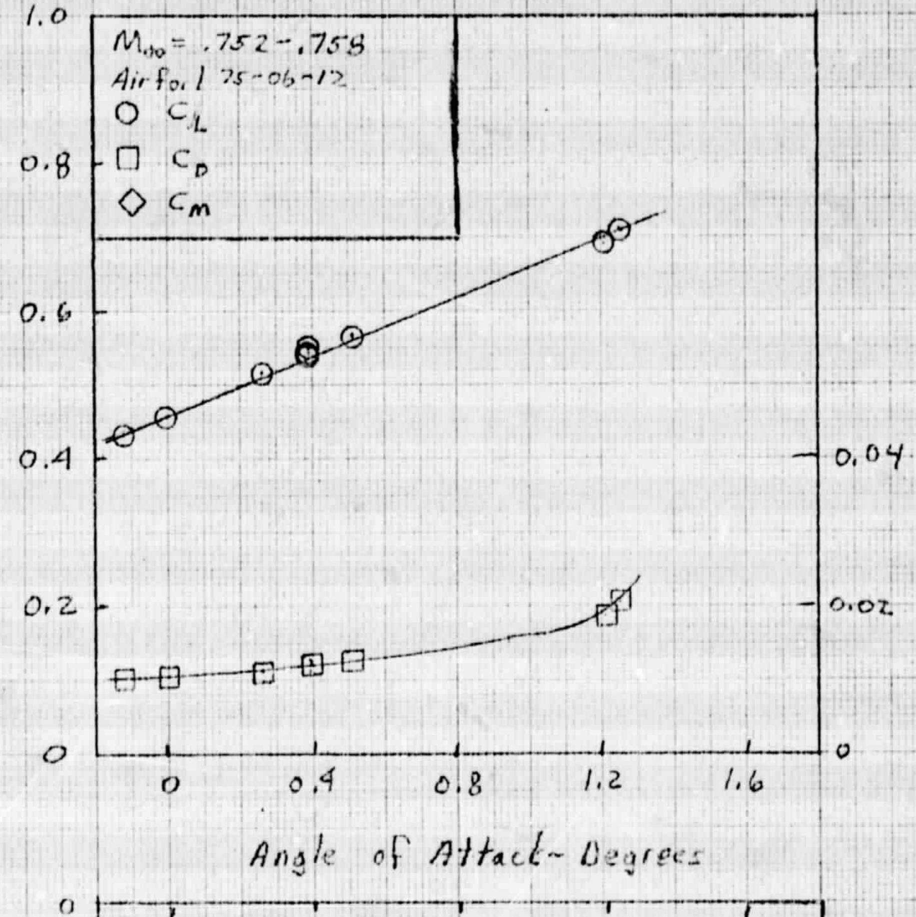

Aerodynamic Coefficients at $M_\infty \approx 0.75$

Figure 10

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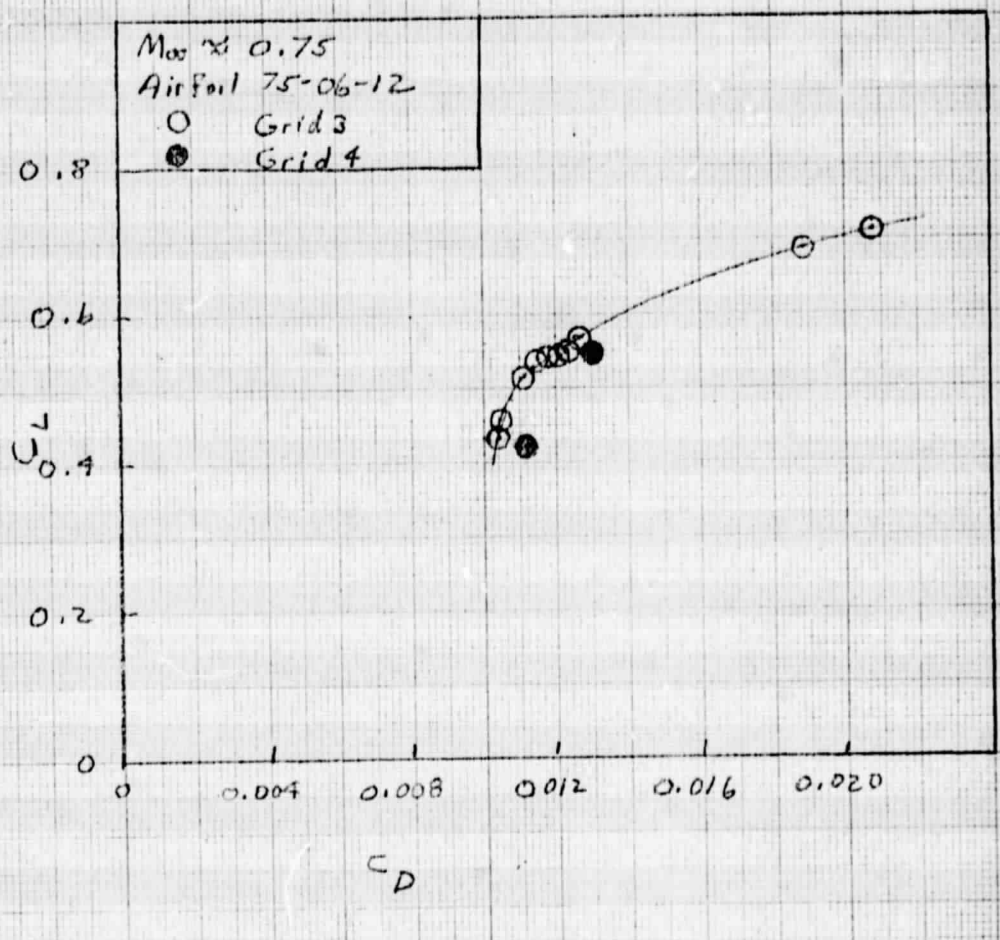

Drag Polar at $M_\infty \approx 0.75$

Figure 11

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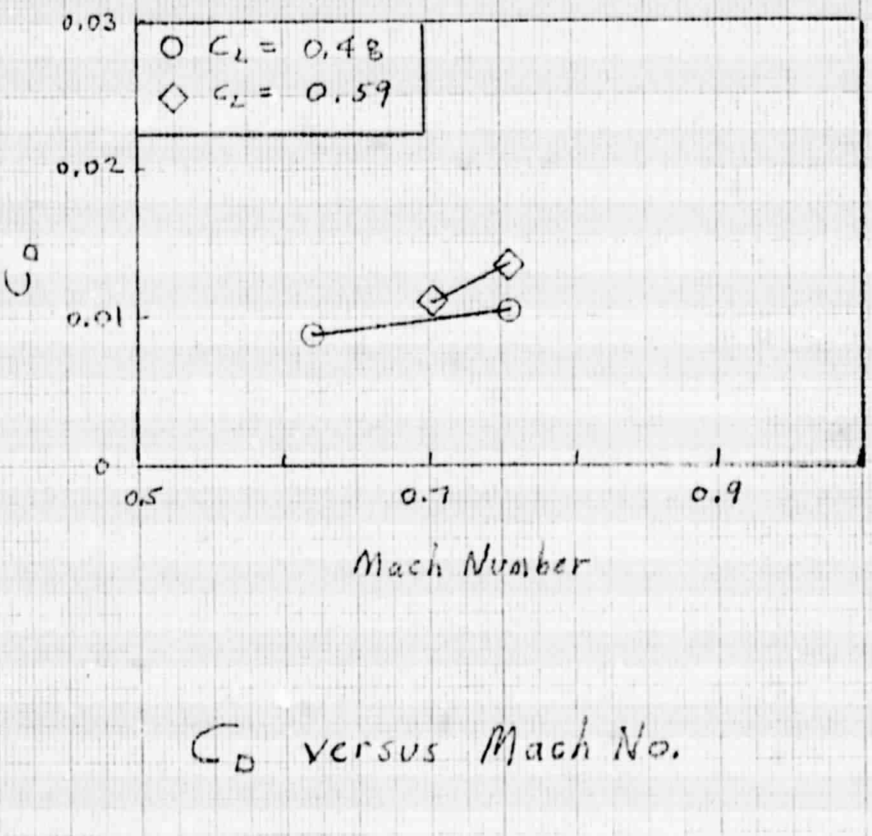

C_D versus Mach No.

Figure 12

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