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NOTATION

c	airfoil chord, cm (in.)
c_d	section drag coefficient
c_D'	local drag coefficient in airfoil wake
c_l	section lift coefficient
c_m	section pitching-moment coefficient referenced to quarter chord
C_p	pressure coefficient $\frac{p_L - p_\infty}{q_\infty}$
C_p^*	pressure coefficient for $M_L = 1$
M	Mach number
p	static pressure, N/m^2 (lb/ft^2)
q	dynamic pressure, N/m^2 (lb/ft^2)
Re	Reynolds number based on free-stream conditions and airfoil chord
T	airfoil thickness, cm (in.)
x	airfoil abscissa, cm (in.)
z	vertical distance in wind tunnel, m(ft)
α	angle of attack, deg

Subscripts:

max	maximum
L	local
∞	free-stream conditions

AN ASSESSMENT OF A MODIFIED POTENTIAL-FLOW CODE FOR CALCULATING THE EFFECT OF SMALL GEOMETRIC CHANGE ON THE PRESSURES AND FORCES OF SUPERCRITICAL AIRFOILS

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SUMMARY

Wind-tunnel test data for two supercritical airfoils has been compared with calculations obtained from a nonconservative, potential flow code over a Mach number range from 0.20 to 0.80. The potential flow code includes an iterated, Nash-McDonald, integral boundary-layer correction. The two supercritical airfoils are closely related with one being derived from the other by making a small modification to the upper surface.

The results of this study showed: good correlation between experimental and theoretical pressure distributions for flow which was entirely subsonic or subsonic with a small supersonic zone; fair correlation between experimental and theoretical pressure distributions for subsonic flow with moderate or greater zones of embedded supersonic flow; fair correlation between experimental and theoretical pitching moments; and poor correlation between experimental and theoretical drag coefficients. The theory did not adequately predict the effect of small geometric change on drag which indicates that accurate gradient information for numerical optimization could not be obtained from the potential flow code evaluated here.

INTRODUCTION

The purpose of this study is to examine the capability of a modified version of a widely used potential flow code, designated program H (ref. 1) to predict the surface pressures and aerodynamic forces on two closely related supercritical airfoil sections. The similarity of the two sections permits an assessment of the ability of Program H to predict the effect of small geometric change on the pressures and forces; a requirement for calculating gradients for design by numerical optimization. Program H solves the two-dimensional, quasi-linear, potential equation

$$(a^2 - u^2)\ \phi xx + (a^2 - v^2)\ \phi yy - 2uv\ \phi xy = 0$$

by successive line overrelaxation using a rotated finite difference scheme to treat both subsonic and supersonic flows. The finite difference scheme is first-order accurate in the supersonic zone and second-order accurate in the subsonic zone. A turbulent boundary-layer displacement thickness is calculated by the Nash-McDonald integral method of reference 2 and added to the airfoil surface at specified times during the relaxation process. This technique is not justified on a purely theoretical basis since the finite difference scheme does not conserve mass when shock waves are present; hence, the streamlines are displaced from the airfoil surface by two separate mechanisms which in effect gives two boundary-layer like corrections. The difference between the original Program H of reference 1 and the code used here is the inclusion in the present code of laminar boundary layer and transition point calculations by the method of Thwaites. The user has the option of specifying the transition point or allowing the program to determine it from the pressure gradients and Reynolds number.

DISCUSSION

The two airfoil sections used during this study are shown in figure 1. Note that the only difference between airfoil SC1 and SC2 is that airfoil SC2 has slightly less thickness over the forward region of the upper surface than SC1. The lower surfaces are identical. The two airfoils were tested in the Ames 2- by 2-Foot Wind Tunnel over a Mach number range from 0.2 to 0.8 with Reynolds numbers varying from 1.9 million at Mach 0.2 to 4 million at Mach numbers greater than or equal to 0.6. Boundary-layer transition was fixed at the 28%-chord station on both surfaces for all test conditions by use of the No. 120 carborundum grit. Data were also obtained with natural transition at selected test conditions. All calculations were performed for a transition location of 0.28 chord. These calculations resulted in a theoretical drag value which was possibly too low for "peaky" pressure distributions. The lift and pitching-moment coefficients were obtained from an integration of surface pressures and the drag coefficients were derived from wake pressure measurements. The theoretical calculations were performed on an 80 × 160 mesh. Data will be shown for 3 lift coefficients for each airfoil. It was not possible to obtain a precise match of lift coefficient for the two airfoils but this is not necessary for the discussion that follows.

The pressure distributions along with tabulated aerodynamic force coefficients for $M = 0.2$ and $Re = 1.9 \times 10^6$ are shown in figure 2. The quantity SEP shown with each pressure distribution determines the separation location in program H and is defined by the equation

$$SEP = \frac{\theta}{u}\frac{du}{ds}$$

where θ is the momentum thickness, u is the local velocity, and s is the arc length measured along the airfoil surface from the stagnation point.

Note that the experimental pressure distributions correlated reasonably well with the theoretical distributions; the only disagreement was the slight shift in the pressure level noted at all lift coefficients. The theoretical calculations were made at the experimental lift coefficient rather than at the experimental angle of attack because of the uncertainty in the wall corrections which were used to correct the geometric angle of attack in the wind tunnel. Such corrections can exceed 2 deg. for high lift conditions. The theoretical and experimental pitching-moment coefficients correlated fairly well, whereas the theoretical drag was consistently below the experimental value. Note also that the experimental drag level of airfoil SC2 was consistently below that of airfoil SC1, whereas theory predicted identical drag for both airfoils. This fact, alone, implied that program H would be difficult to use as a design code with a numerical optimization algorithm that was based on gradient information. Program H indicated upper surface, trailing-edge separation at all lift coefficients. No attempt was made to verify this prediction during testing; however, the trailing-edge pressure coefficient indicated the possibility of mild separation near the trailing edge.

The pressure distributions and aerodynamic force coefficients for $M = 0.40$ and $Re = 3 \times 10^6$ are shown in figure 3. The main difference between the experiment-theory correlation shown in figure 3 and that of figure 2 is that the slight pressure level shift noted at $M = 0.20$ has disappeared at $M = 0.4$ giving better agreement between experimental and theoretical pressure distributions at the higher Mach number. The degree of correlation for the aerodynamic force coefficients is similar to that shown at $M = 0.20$. Again, Program H predicts the same level of drag for both airfoils, whereas experiment shows a lower drag level for Airfoil SC2 at all lift coefficients.

The pressure distributions and aerodynamic force coefficients for $M = 0.60$ and $Re = 4 \times 10^6$ are shown in figure 4. The aerodynamic force coefficients are shown for Airfoil SC1 with free transition at all lift coefficients to show the effect of roughness on the pressures and forces. Note that the pressures with fixed transition are nearly identical to those with free transition and as expected the drag coefficients are affected by roughness. Note, also, that the theory reproduces the supersonic zone near the upper-surface leading edge very well (figs. 4(e) and 4(f)).

The pressure distributions and force coefficients for $M = 0.70$ and $Re = 4 \times 10^6$ are shown in figure 5. The degree of correlation between experimental and theoretical pressure distributions is not as good as found at lower Mach numbers (figs. 2, 3, and 4). In particular, the pressures over the forward region of the upper surface at the highest lift coefficients are not predicted well (figs. 5(3) and 5(f)). It appears that good experiment-theory correlation of pressures is obtained for subsonic flow or for mixed subsonic-supersonic flow if the supersonic flow is confined to the forward 2-3% of the chord. Some of the discrepancy can be attributed to a lack of consideration of the shock boundary-layer interaction in program H. Note also that at this Mach number, experiment shows airfoil SC2 to have the lower drag whereas theory shows the opposite trend.

The pressure and force data for $M = 0.75$ shown in figure 6 indicate somewhat better agreement between experiment and theory than at $M = 0.70$ (fig. 5). In particular, the pressure correlation at the highest lift coefficients (figs. 6(e) and 6(f)) is better than that for $M = 0.7$ (figs. 5(e) and 5(f)). It appears that as the shock moves aft into the turbulent boundary-layer region, better experiment/theory correlation is achieved. Matching experimental and theoretical lift coefficients appear to cause theoretical pressures which are more positive ahead of the shock and more negative behind the shock for many test conditions (e.g., fig. 5(e)). Proper theoretical treatment of the shock boundary-layer interaction might reduce such discrepancy.

At $M = 0.76$ the correlation is better at the higher lift coefficients (figs. 7(e) and 7(f)) than at the intermediate lift coefficients (figs. 7(c) and 7(d)). The theory did not capture a weak shock for airfoil SC2 (fig. 7(d)) on the 80×160 mesh. A calculation of a finer mesh might have captured the shock, but time and money precluded such a calculation. It is also possible that the position and strength of the experimental shock is influenced by a grit-generated disturbance which is not modeled by program H. Note that the upper surface separation point is predicted to be somewhat farther forward than that for the lower Mach numbers; a condition that is not consistent with the trailing-edge pressure recovery shown in figures 7(e) and 7(f).

The data for $M = 0.77$ shown in figure 8 exhibit a similar degree of correlation between experiment and theory as seen for $M = 0.76$. Again, the weak shock shown in the experimental data of figure 8(d) is not captured by the theory. Note that for this Mach number, experiment and theory agree as to which airfoil has the lower drag at all lift coefficients. However, this is a fortuitous result because the data for M = 0.78 (the design Mach number), shown in figure 9 again show the lower theoretical drag for airfoil SC1 at the lowest lift coefficients (figs. 9(a) and 9(b)). Furthermore, the data for M = 0.79 and M = 0.80 shown in figures 10 and 11, respectively show a lower theoretical drag for airfoil SC1 at nearly all lift coefficints which is opposite to the experimental result. It appears that Program H, in its present form, may have a drag coefficient accuracy of approximately ±0.0015 which means the code cannot consistently predict the effect of small geometric change on the drag coefficient. Because of this inconsistency, drag minimization problems would be very difficult. There are several possible explanations for this randomness in the drag calculations. First, the boundary-layer displacement thickness calculated by the Nash-McDonald method is rough and must be smoothed before adding it to the airfoil. Such smoothing causes some randomness, particularly in the trailing-edge quantities which are used in Program H with the Squire-Young formula for calculating form drag. Another source of error is the integration of surface pressures to obtain wave drag. Such integration is known to be inaccurate because of a finite number of pressures available. The inviscid relaxation process may be yet another source of error. In most cases the process is terminated when the maximum residual has been reduced several orders of magnitude, but is still well above that attainable with the CDC 7600 which was used during this study. Furthermore, the finite difference equations have inherent truncation errors which cannot be totally eliminated. It is not uncommon to find a wave-drag coefficient of ±0.0010 for purely subsonic flow from program H calculations. Another source of error is the lack of wake treatment in the Nash MacDonald method.

Wake profiles are shown in figure 12 for Mach numbers of 0.78, 0.79 and 0.80. These profiles are typical of the type of data used to determine the drag coefficients tabulated in figures 2 through 10. It seems clear from the wake profiles that the data are sufficiently smooth to preclude attributing the disagreement between experimental and theoretical drag levels to inaccuracies in the wind-tunnel measurements.

CONCLUDING REMARKS

An experiment-theory correlation study was conducted to assess the capability of a widely used transonic potential code to analyze and design transonic airfoils. The following results were established.

1. The drag calculations are not sufficiently consistent to permit the code to be used with a gradient optimization algorithm for drag minimization problems.

2. The theoretical pressure distributions agree well with the experimental distributions for purely subsonic flow and mixed subsonic-supersonic flow if the supersonic zone is confined to the forward 2 or 3% of the chord.

3. The correlation between the experimental and theoretical pressure distributions is least satisfactory when the shock is located near the 20% chord station. This may be related to the formation of a separation bubble at the base of the shock which is neglected by the theory.

4. Weak shocks may not be captured by the theory for an 80 × 160 mesh.

5. The separation point is not adequately predicted by theory.

6. The theoretical chordwise location of moderate strength shocks usually agrees to within 10% chord of the experimental position.

7. The general shape of the pressure distributions was predicted fairly well by the theory for most test conditions.

8. The theoretical pitching-moment coefficients agreed fairly well with the experimental values at most test conditions.

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1. Bauer, Frances; Garabedian, Paul; Korn, David; and Jameson, Antony: Supercritical Wing Sections II, Lecture Notes in Economies and Mathematical Systems. Springer-Verlag, 1975.

2. Nash, J. F.; and MacDonald, A.G.J.: The calculation of Momentum Thickness in a Turbulent Boundary Layer at Mach Numbers up to Unity. Aeronautical Research Council C.P. No. 963, 1967.

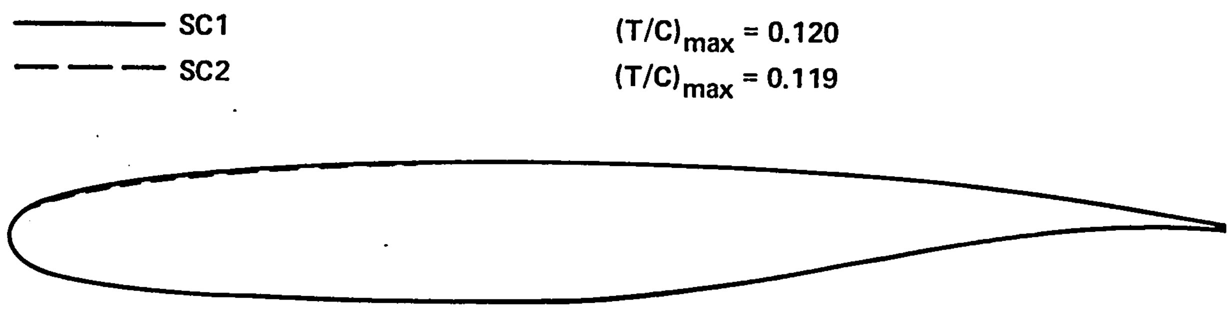


Figure 1.- Airfoil geometry comparison.

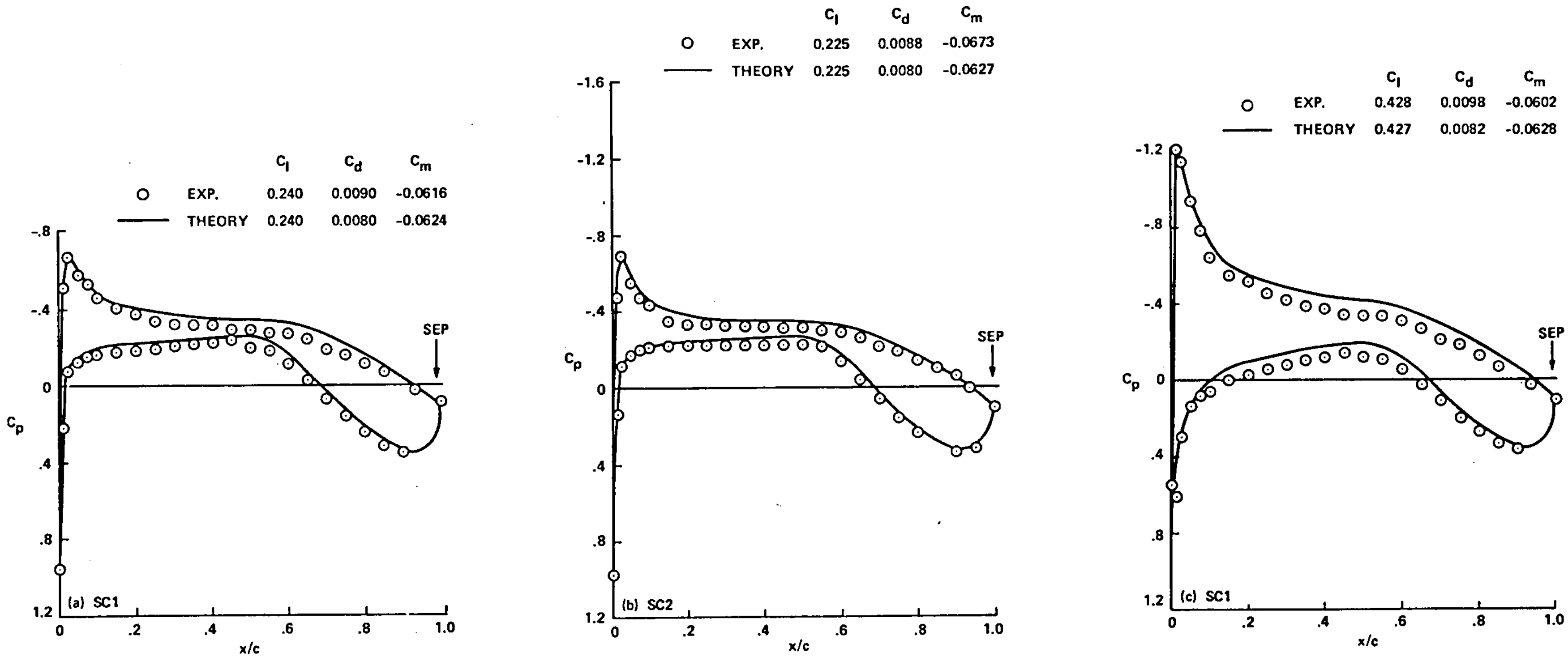
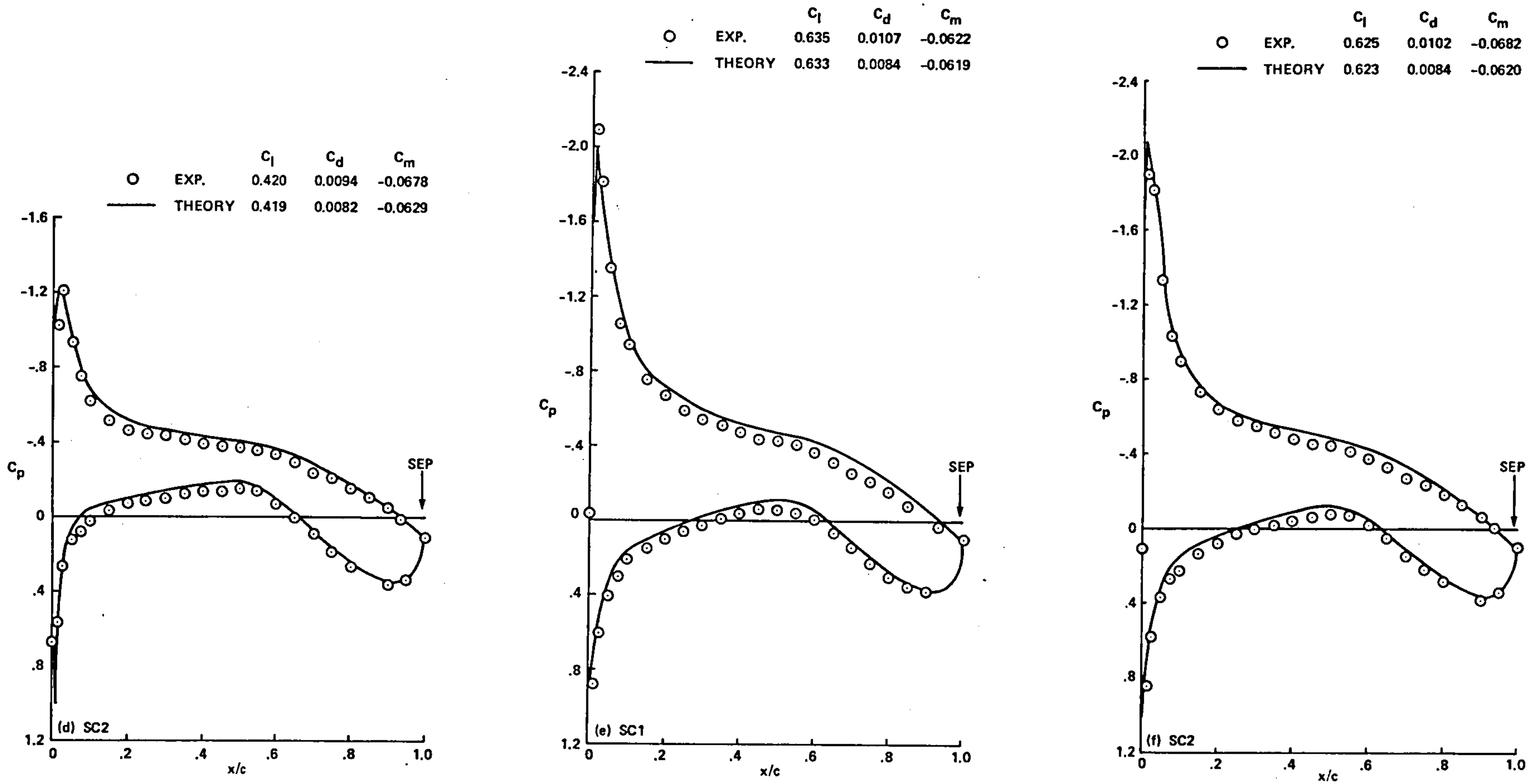


Figure 2.- Pressure distributions, M = 0.2, Re = 1.9 × 10^6, transition fixed.



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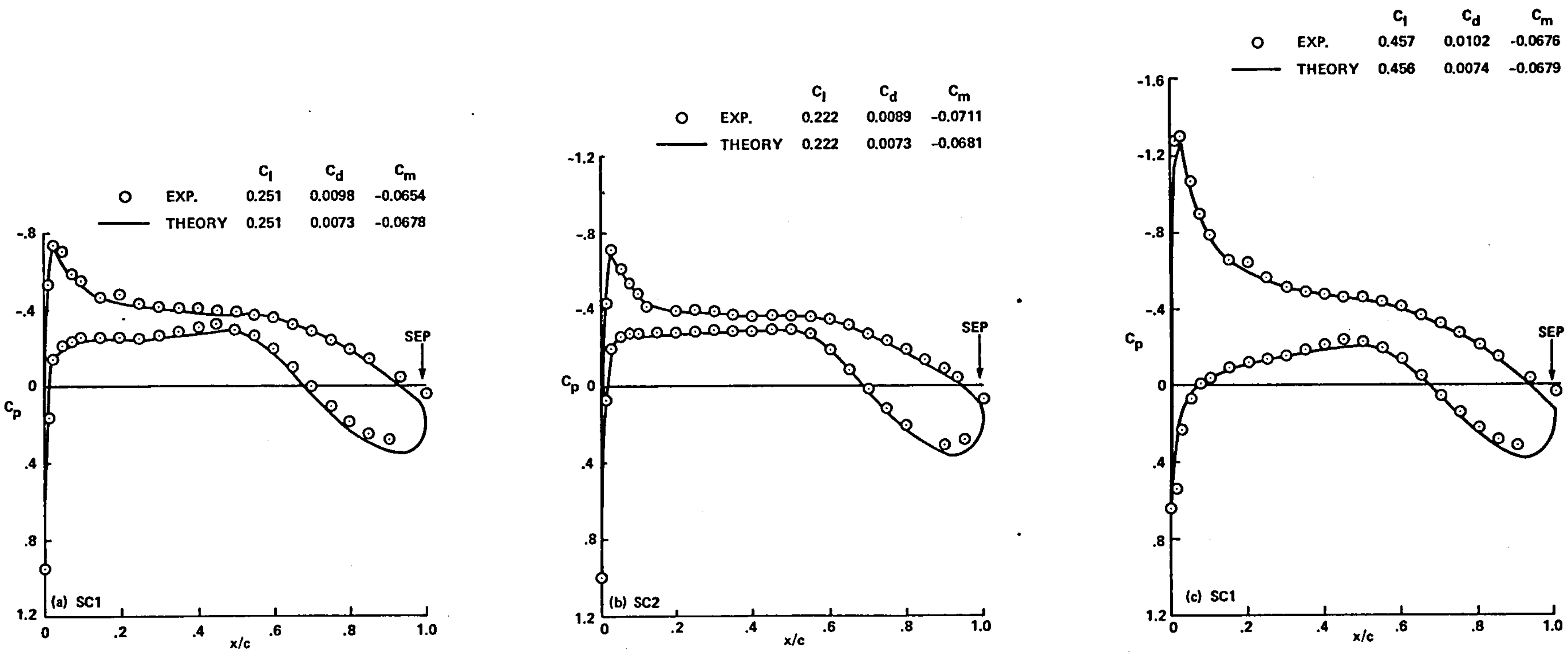
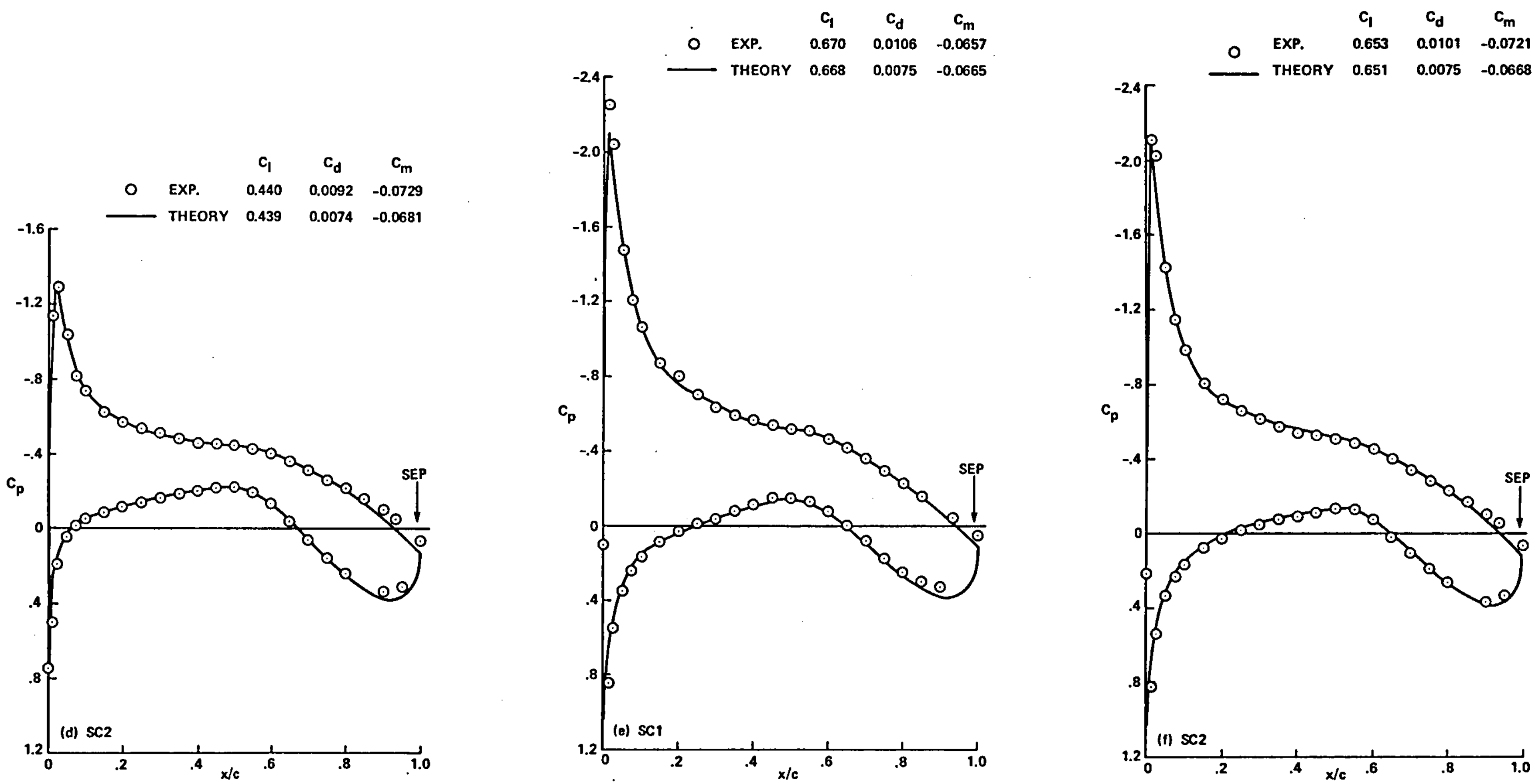


Figure 3.- Pressure distributions, M = 0.40, Re = 3.0×10^6, transition fixed.



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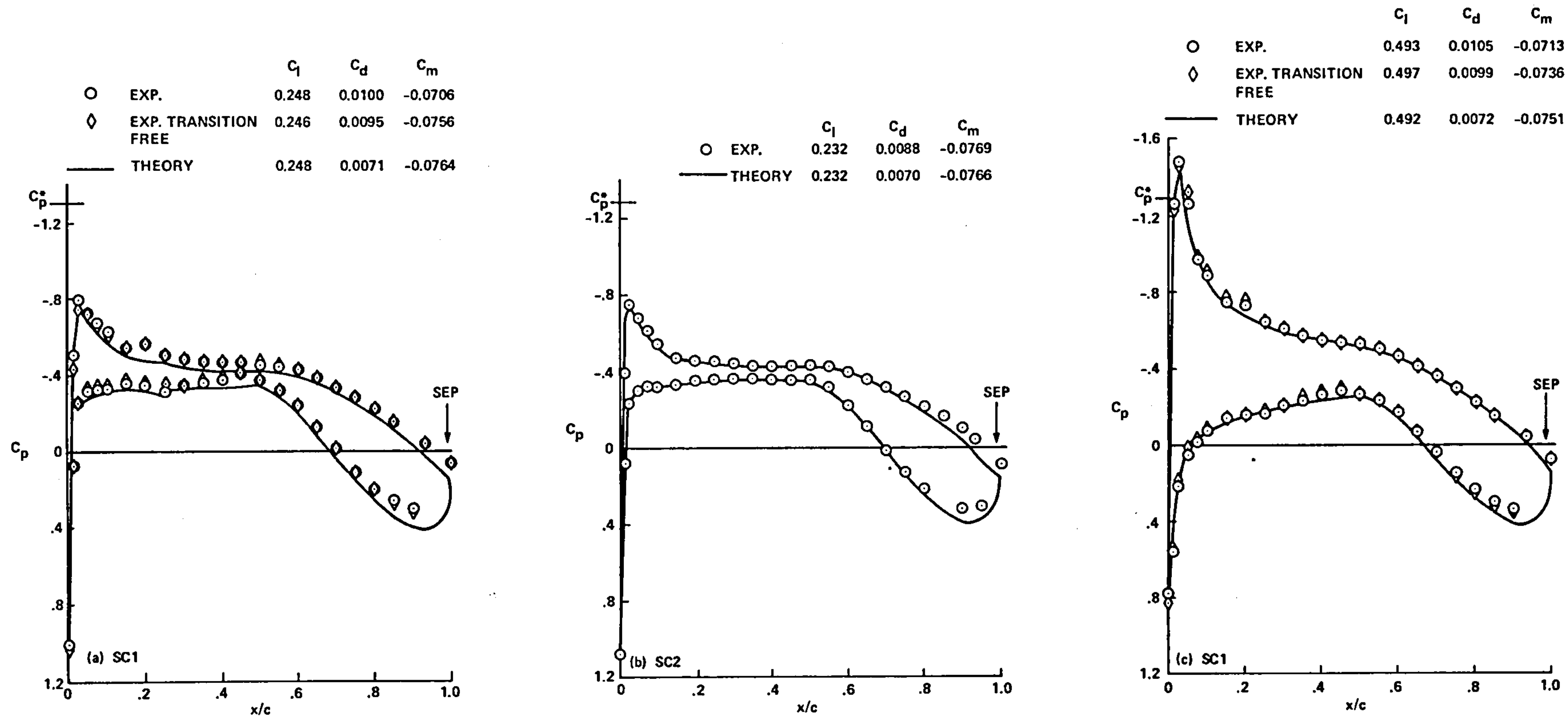
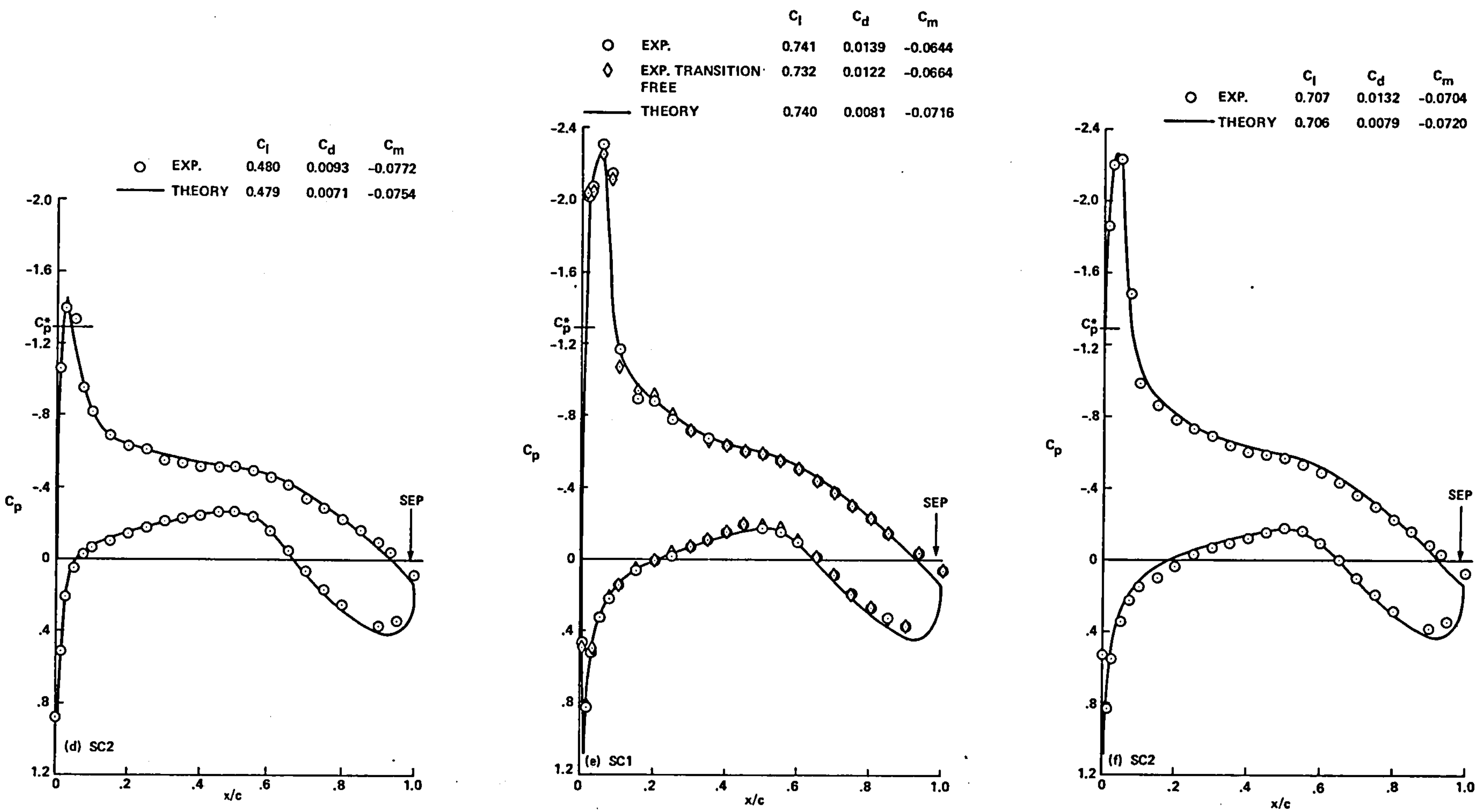


Figure 4.- Pressure distributions, M = 0.60, Re = 4.0 × 10^6, transition fixed except as noted.



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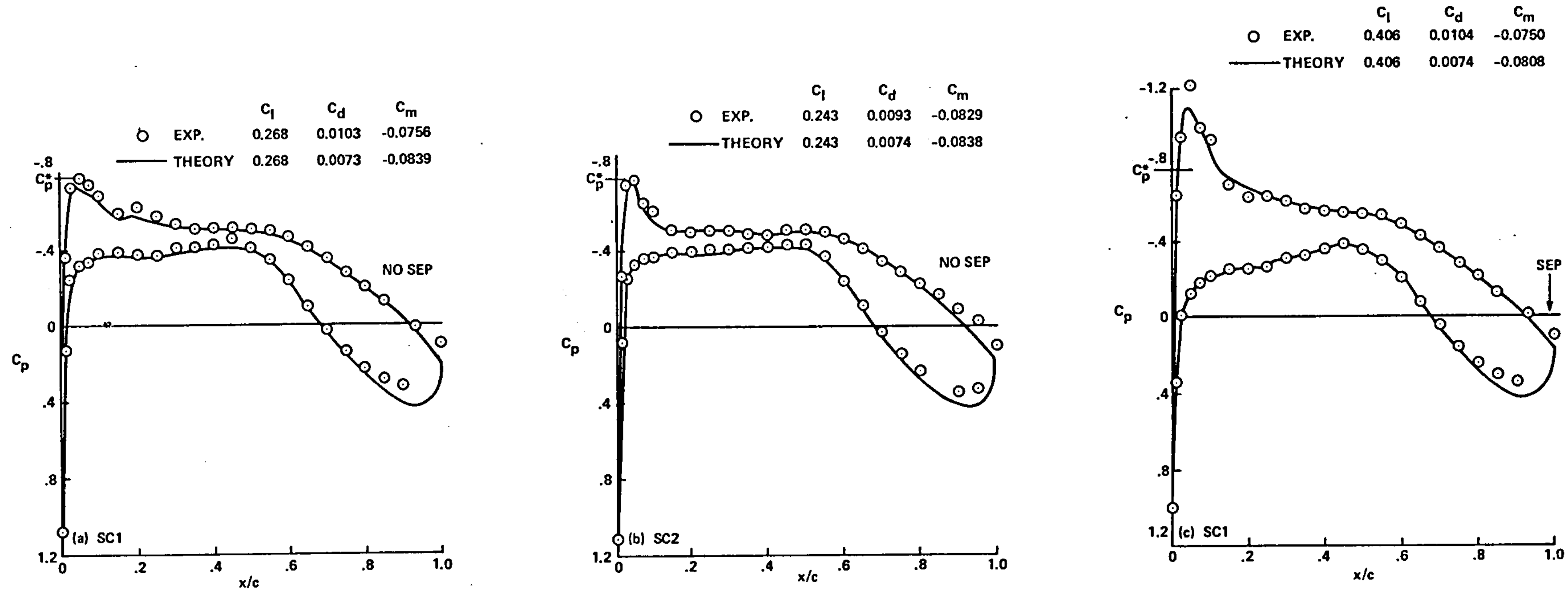


Figure 5.- Pressure distributions, M = 0.70, Re = 4.0 × 10^6, transition fixed.

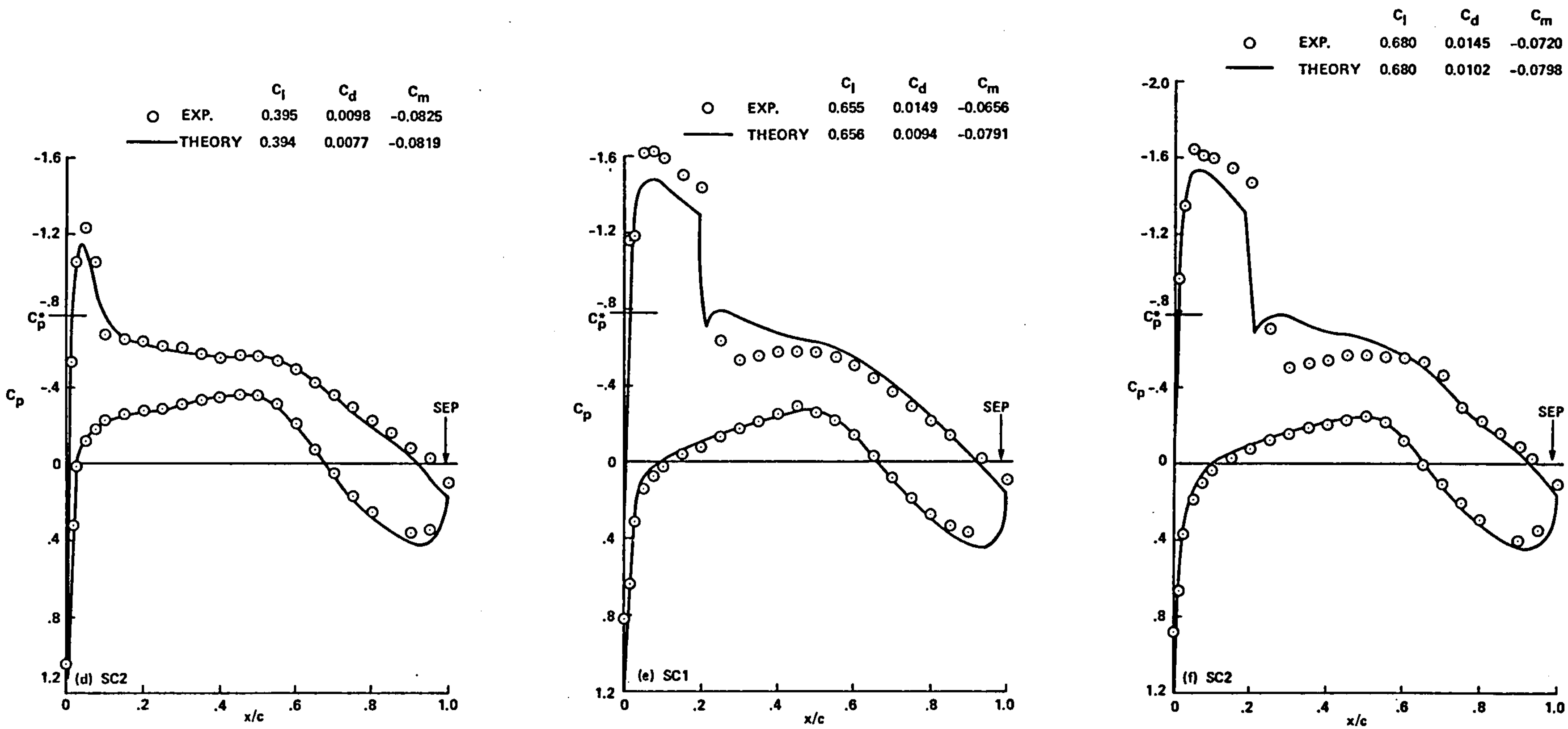


Figure 5.- Concluded.

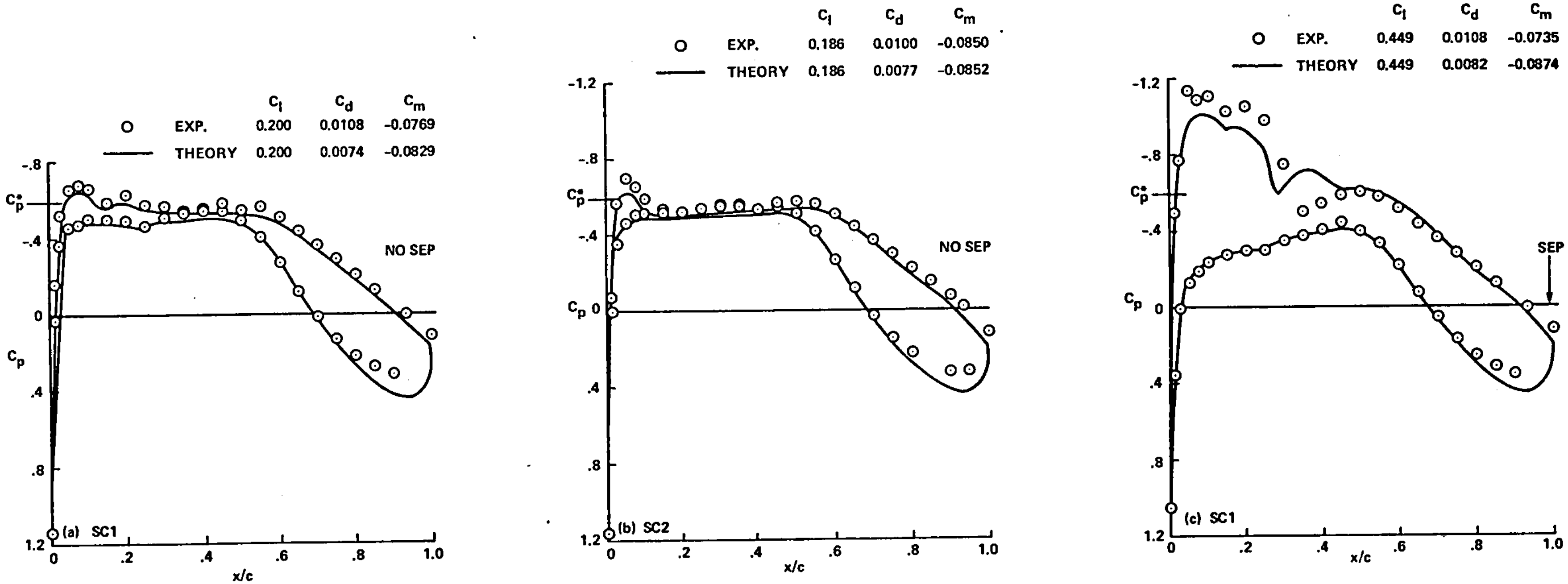
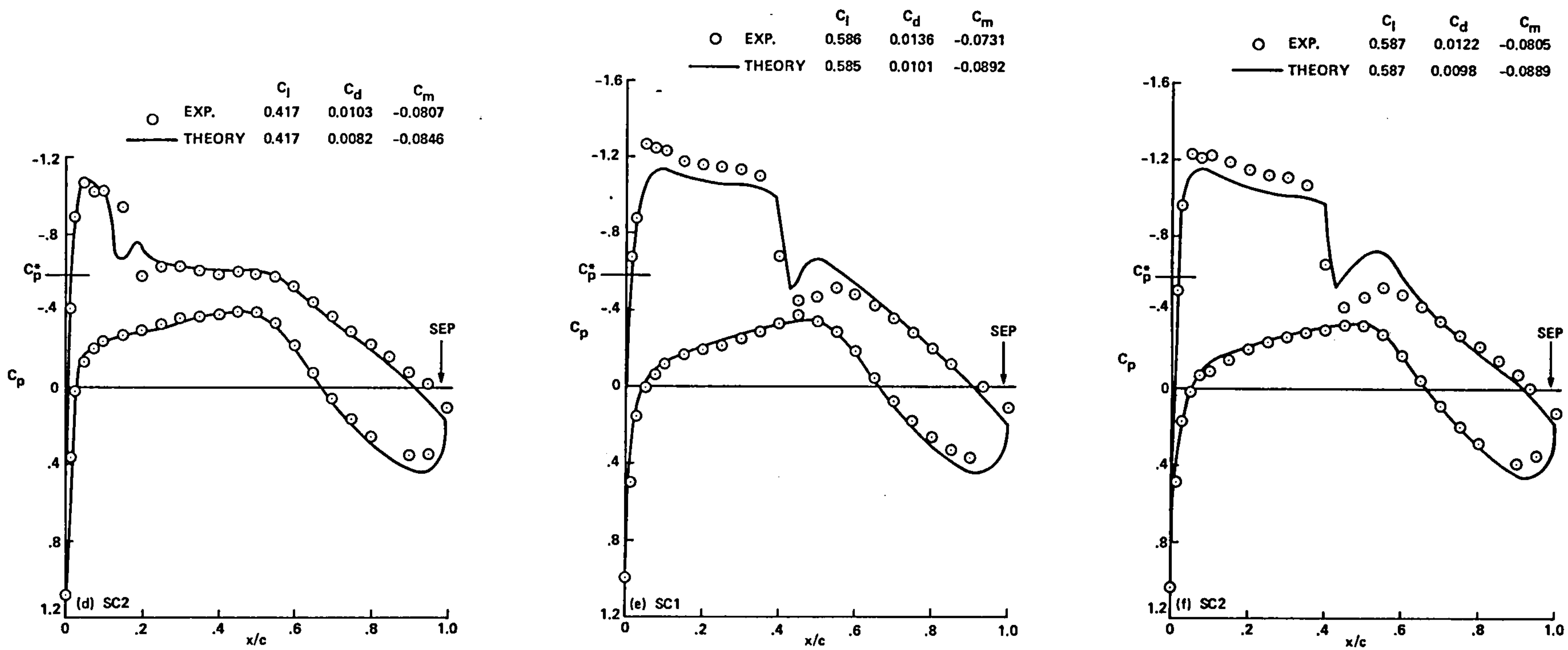


Figure 6.- Pressure distributions, M = 0.75, Re = 4.0 × 10^6, transition fixed.



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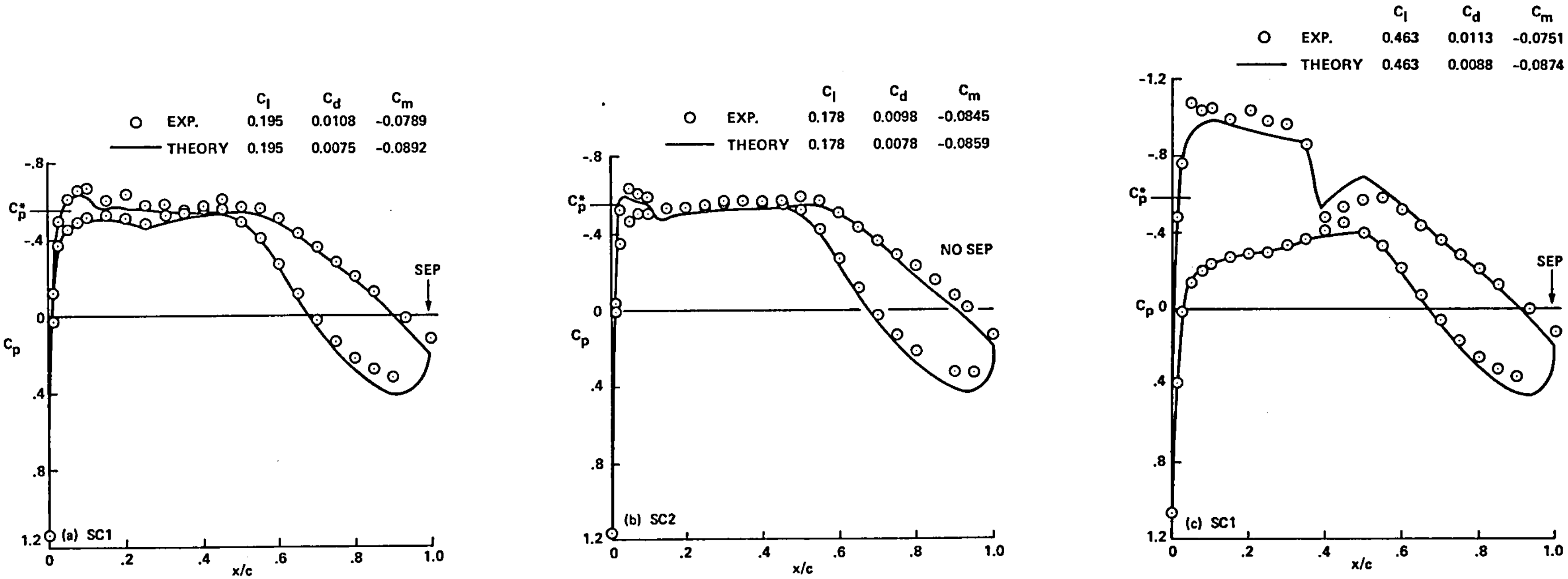
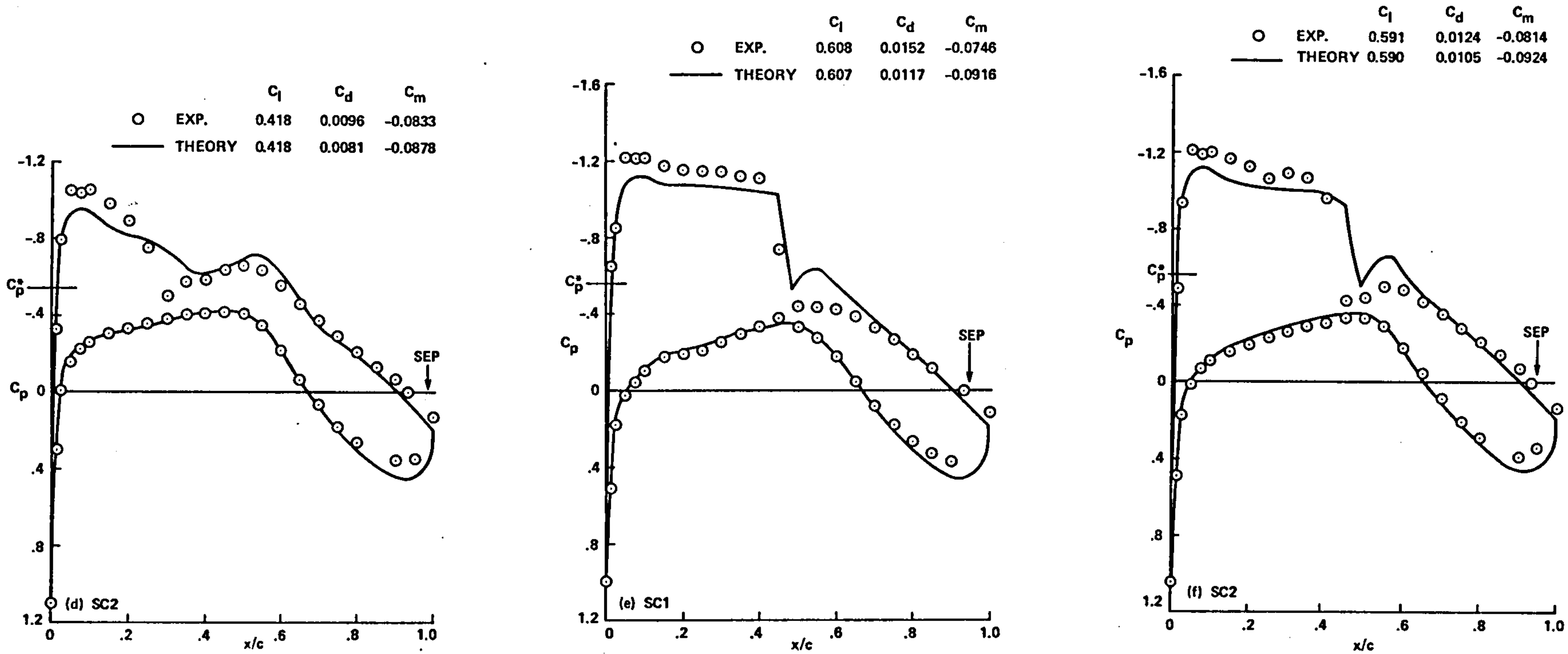


Figure 7.- Pressure distributions, M = 0.76, Re = 4.0 × 10^6, transition fixed.



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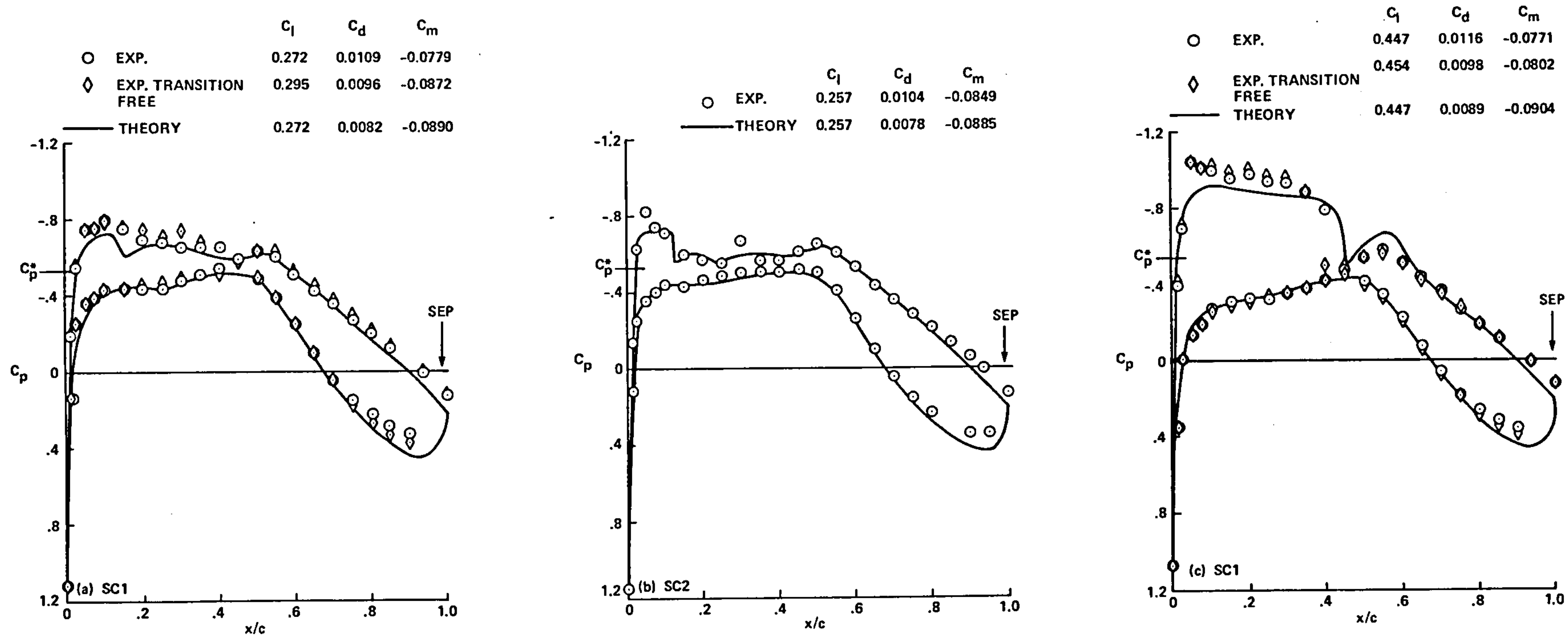
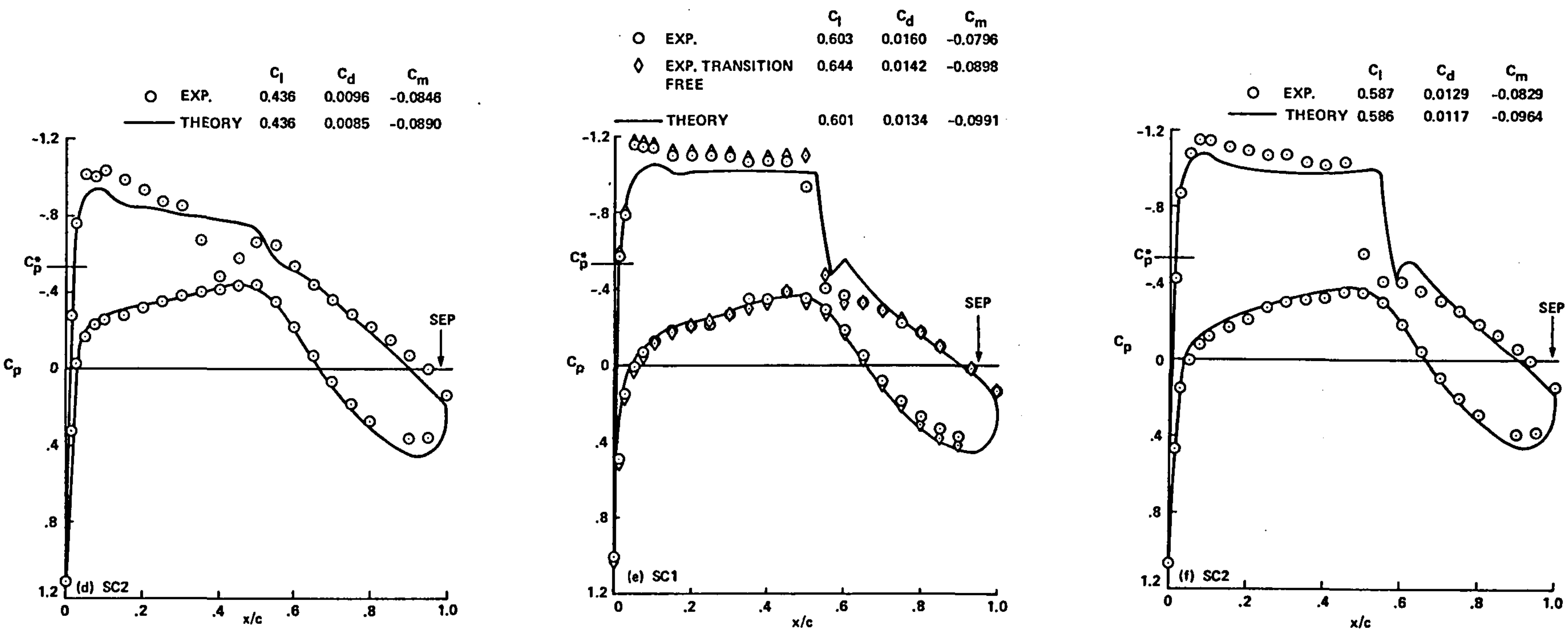


Figure 8.- Pressure distributions, M = 0.77, Re = 4.0×10^6, transition except as noted.



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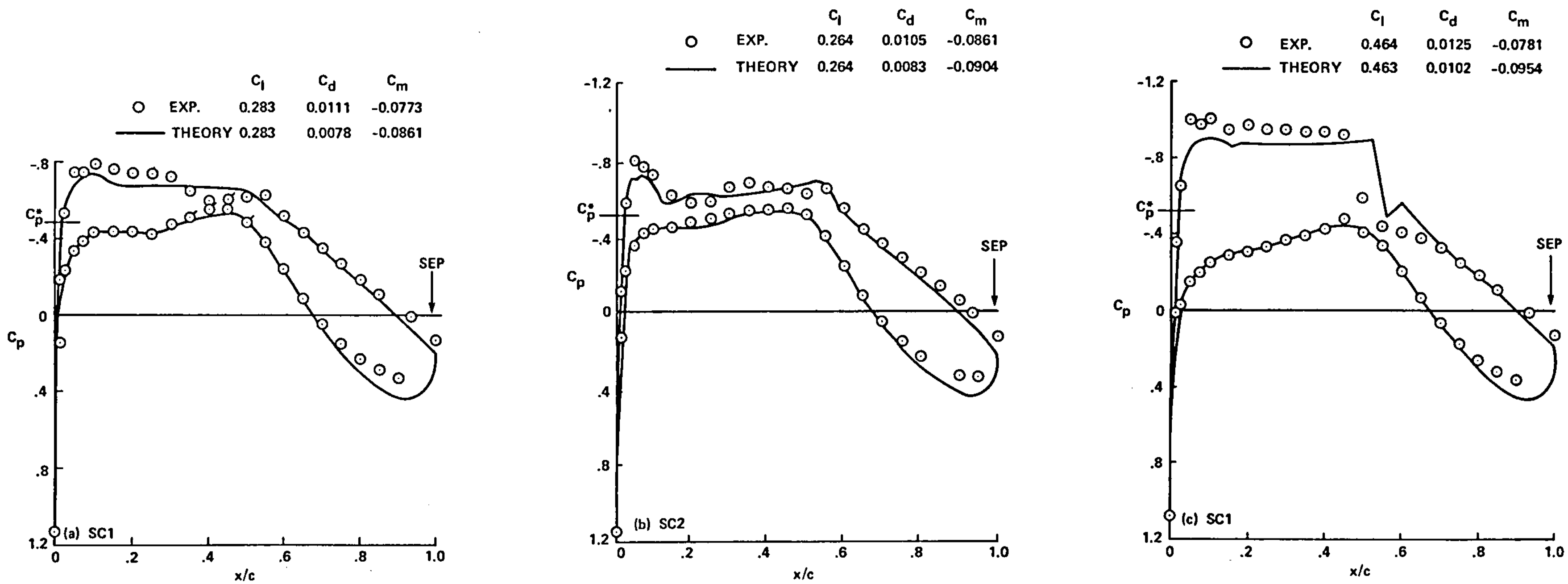
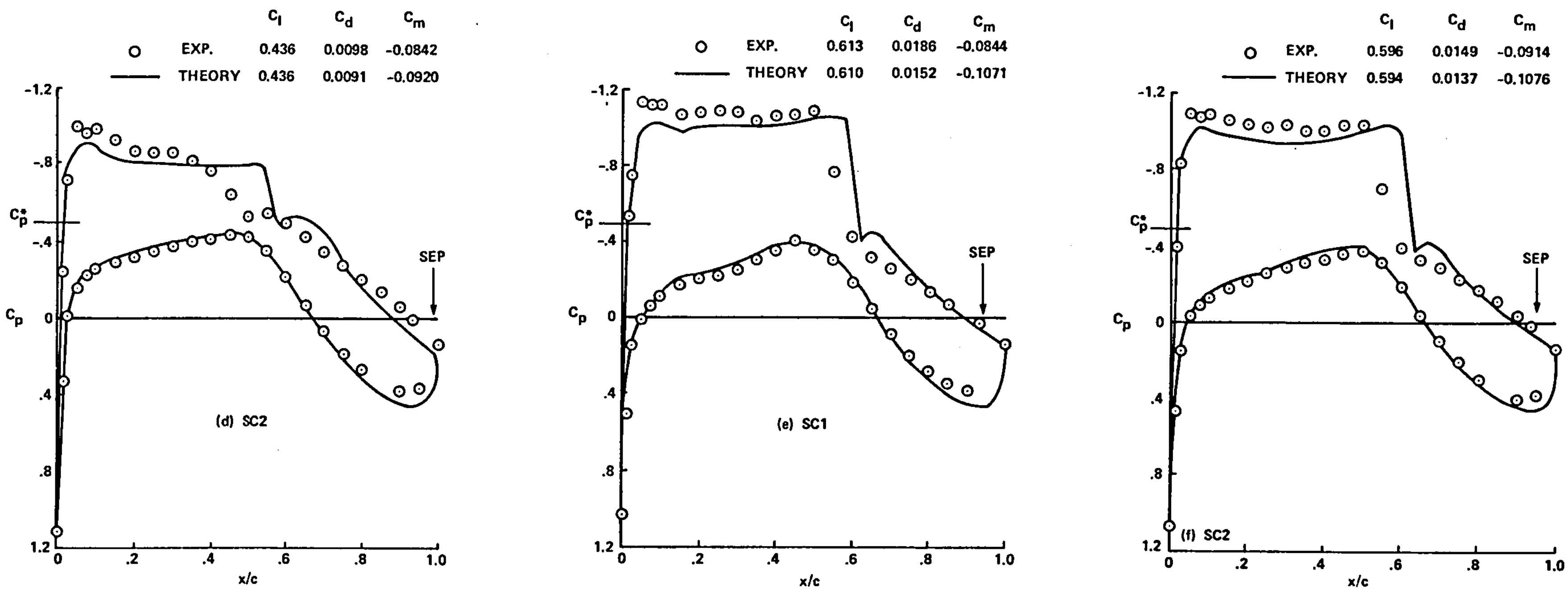


Figure 9.- Pressure distributions, M = 0.78, Re = 4.0 × 10^6, transition fixed.



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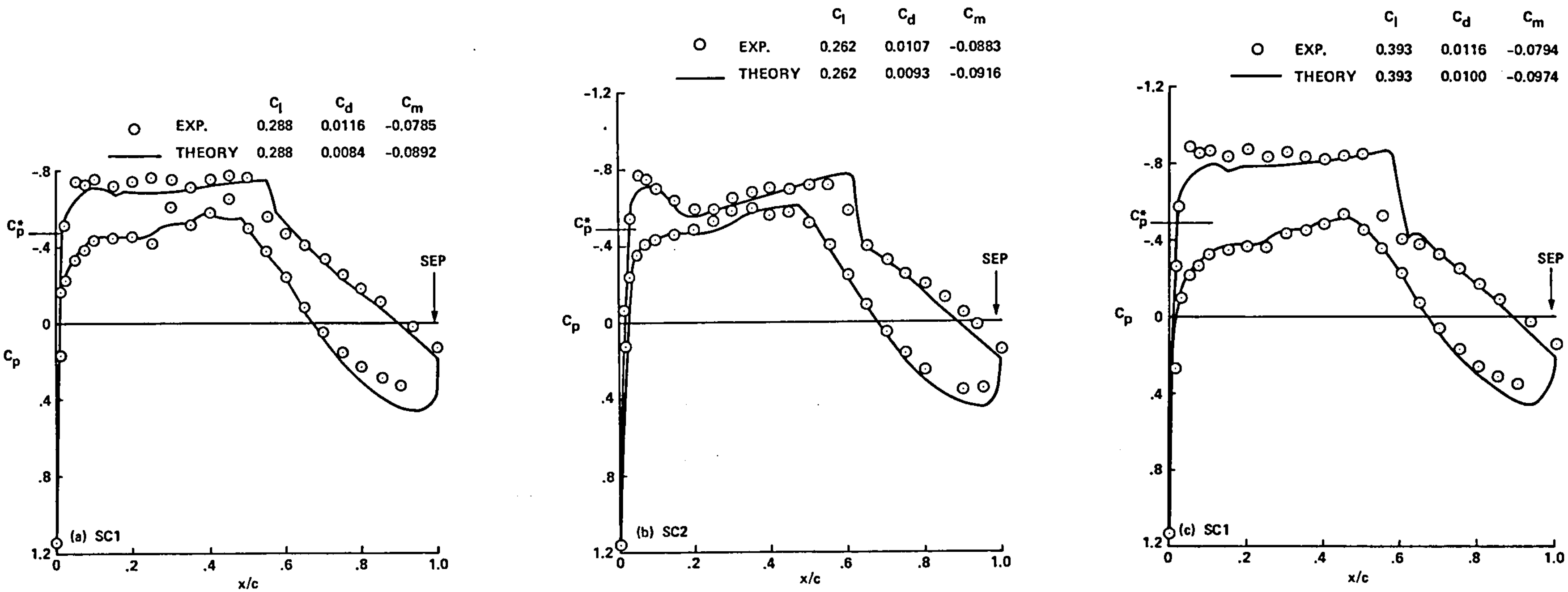
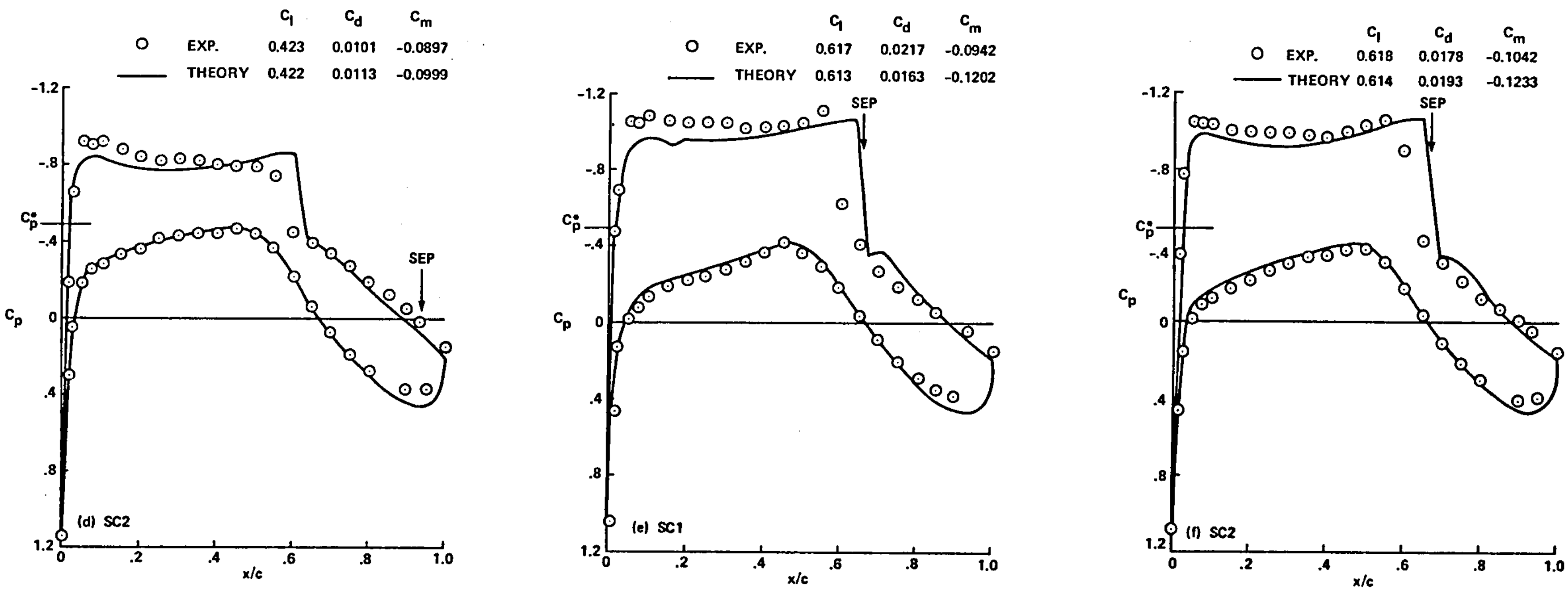


Figure 10.- Pressure distributions, M = 0.79, Re = 4.0 × 10^6, transition fixed.



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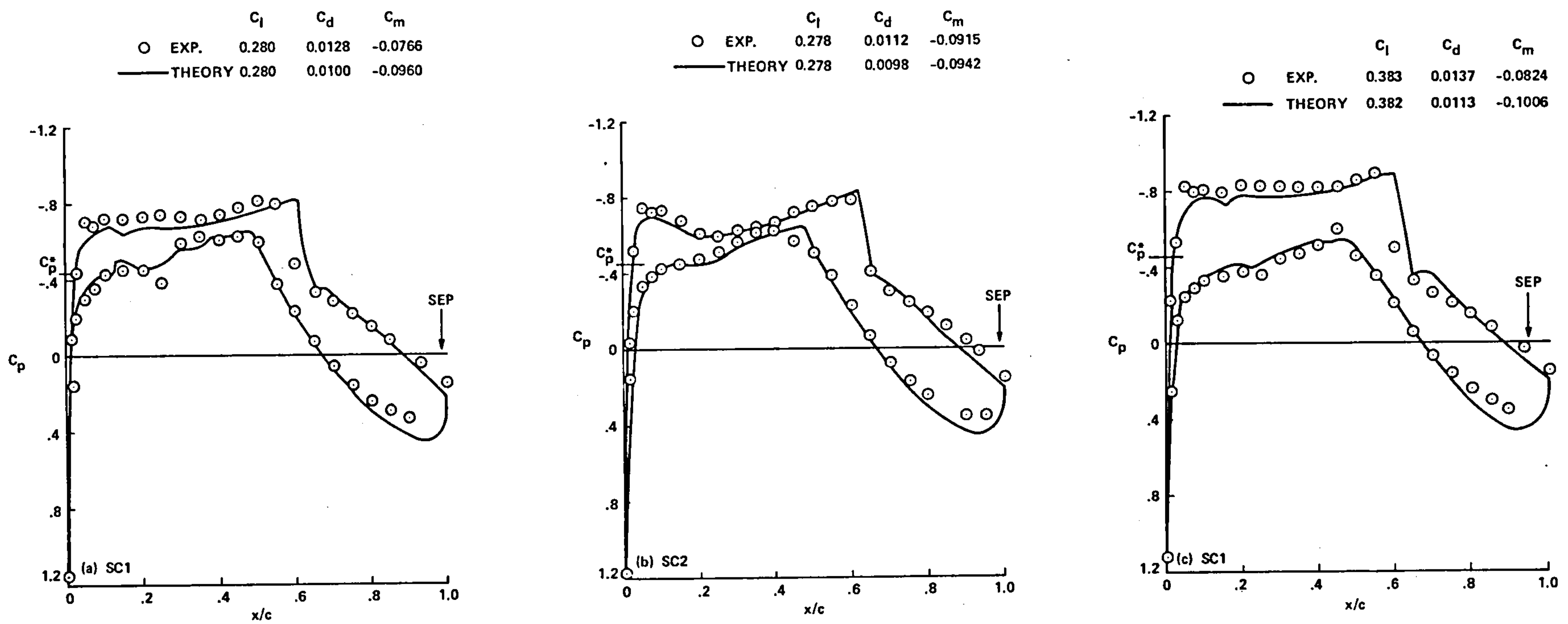
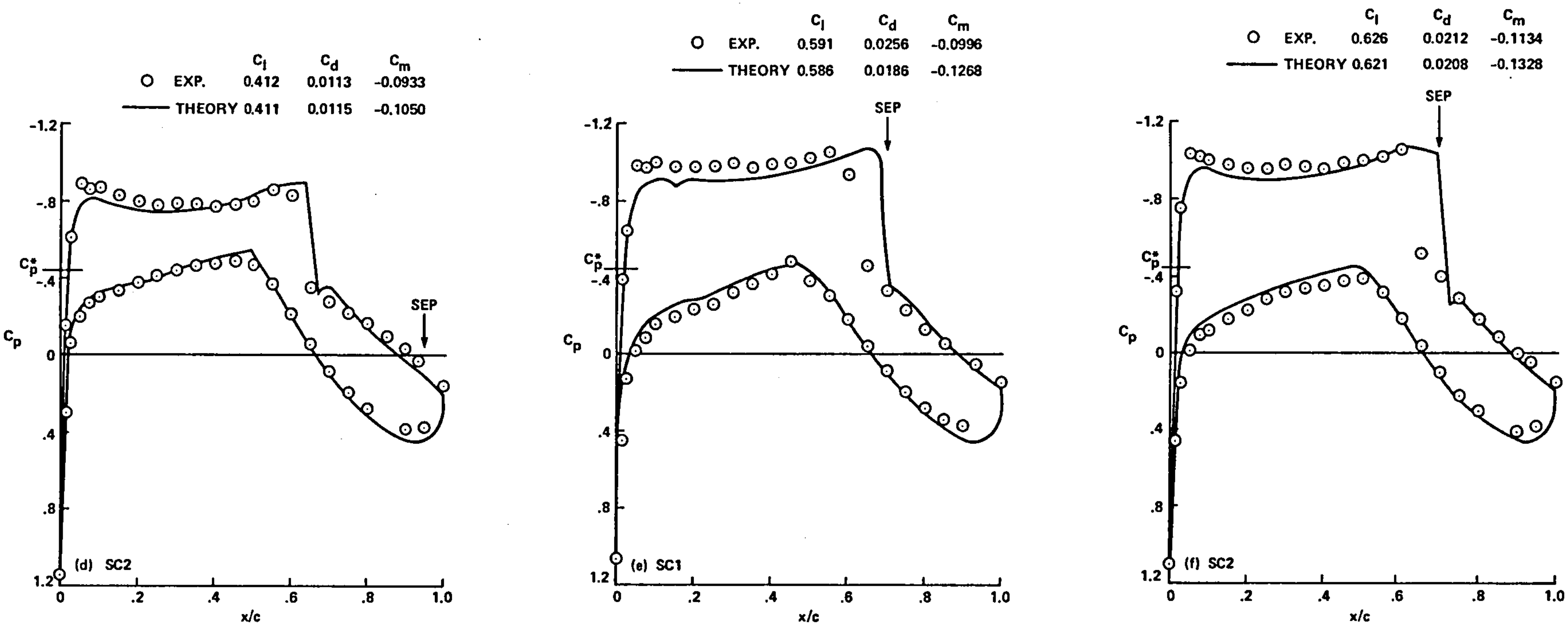


Figure 11.- Pressure distributions, M = 0.80, Re = 4.0×10^6, transition fixed.



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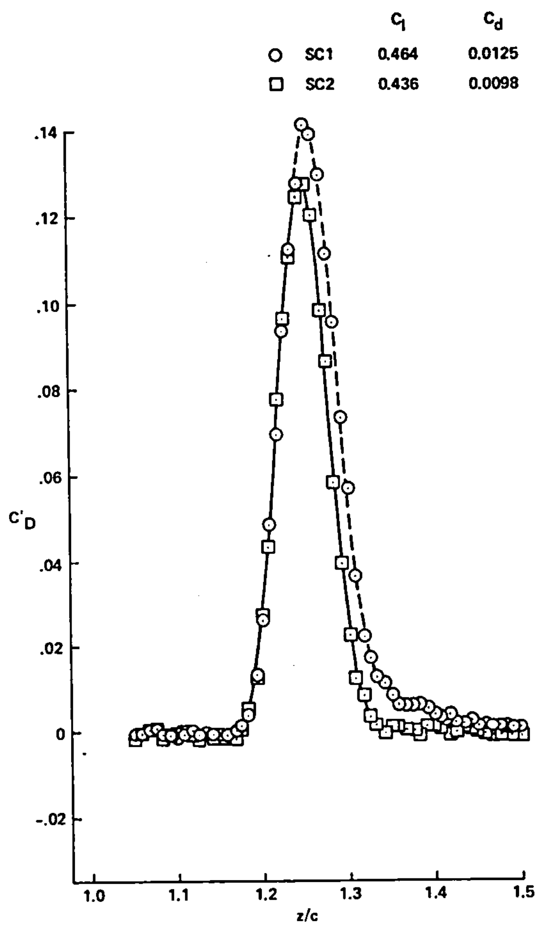

Figure 12.- Wake profiles, M = 0.78, Re = 4.0 × 10^6.

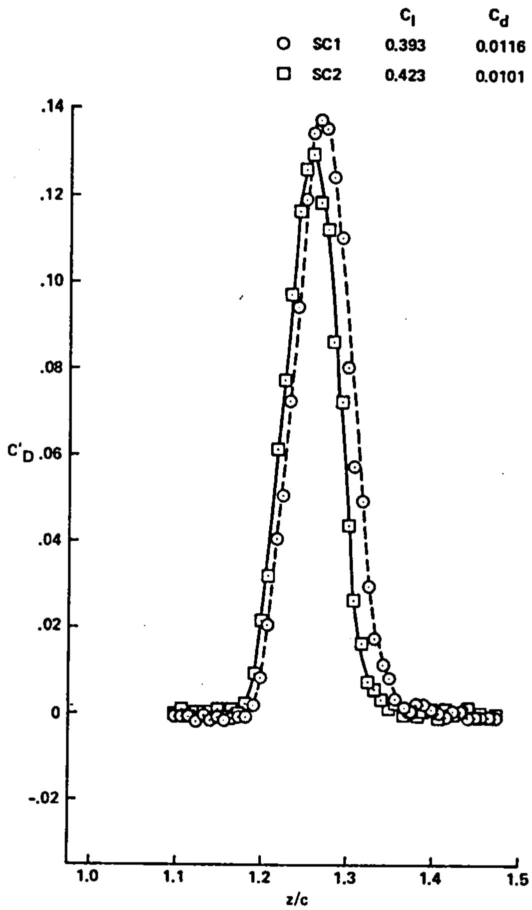

Figure 13.- Wake profiles, M = 0.79, Re = 4.0 × 10^6.

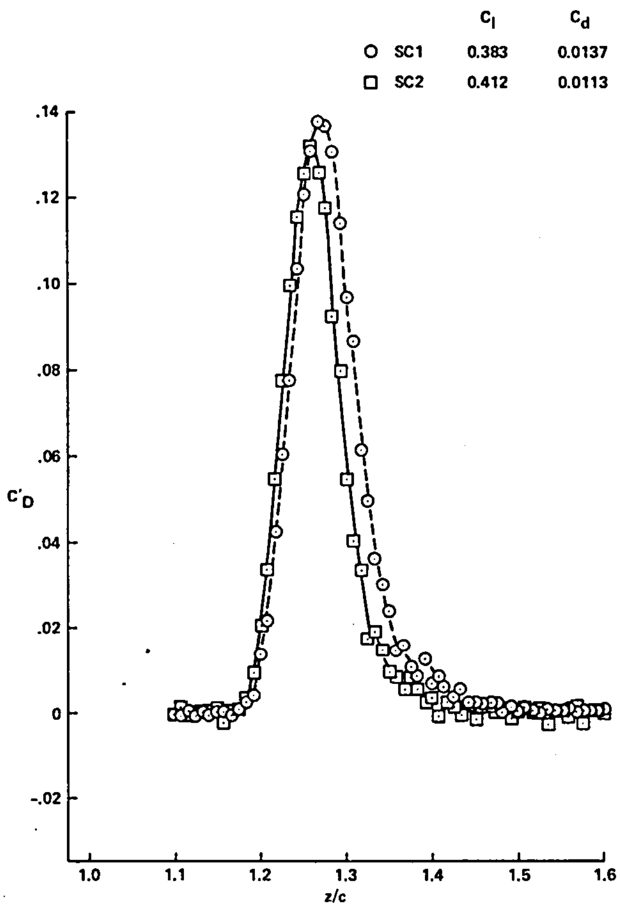

Figure 14.- Wake profiles, M = 0.80, Re = 4.0×10^6.

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15. Supplementary Notes

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16. Abstract

Wind-tunnel test data for two closely related supercritical airfoils has been compared with calculations obtained from a nonconservative, potential flow code over a Mach number range from 0.20 to 0.80. The potential flow code includes an iterated, integral boundary-layer correction.

The results of this study indicate that the theoretical pressure distributions correlated more closely with the experimental pressure distributions when the flow was entirely subsonic or subsonic with a small supersonic zone than when the flow contained a large supersonic zone. The predicted drag level was below the experimental values at nearly all test conditions and the difference in drag level for the two airfoils was not accurately predicted.

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