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COMPARISON OF THE EXPERIMENTAL AERODYNAMIC CHARACTERISTICS OF THEORETICALLY AND EXPERIMENTALLY DESIGNED SUPERCRITICAL AIRFOILS

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16. Abstract

A lifting airfoil theoretically designed for shockless supercritical flow utilizing a complex hodograph method has been evaluated in the Langley 8-foot transonic pressure tunnel at design and off-design conditions. The experimental results are presented and compared with those of an experimentally designed supercritical airfoil which were obtained in the same tunnel.

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COMPARISON OF THE EXPERIMENTAL AERODYNAMIC CHARACTERISTICS OF THEORETICALLY AND EXPERIMENTALLY DESIGNED SUPERCRITICAL AIRFOILS*

By Charles D. Harris
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SUMMARY

Experimental data have been obtained on a 10-percent-thick lifting airfoil theoretically designed for shockless supercritical flow at a Mach number of 0.78 and a lift coefficient of 0.59 by utilizing a complex hodograph method. The data are compared with the experimental aerodynamic characteristics of an experimentally designed supercritical airfoil. At near-design conditions, agreement between the experimental aerodynamic characteristics of the two airfoils was good. Discrepancies in off-design characteristics indicate the necessity for evaluating off-design characteristics as part of the design process in order to realize the full potential of the complex hodograph method as a design tool. The results further suggest that a trade-off between minimum wave losses at the supercritical design point and acceptable off-design characteristics would be a more realistic design goal than a single-point shockless design goal.

INTRODUCTION

With the evolution of advanced technology aircraft (ref. 1), cruise speeds have been extended well into the supercritical speed range where extensive regions of supersonic flow exist on the wings. This evolution, combined with recent successes in achieving virtually shock-free flow in wind-tunnel tests of two-dimensional airfoils (for example, refs. 2 and 3), has given impetus to the development of a practical approach to the theoretical design of transonic lifting airfoils with minimum wave losses.

One approach has been the complex hodograph method for the design of shockless supercritical airfoils reported in reference 4. This mathematical approach was used by P. Garabedian of New York University to design an airfoil to be shock free (isentropic recompression) at a Mach number of 0.78, a lift coefficient of 0.59 and with a maximum thickness-chord ratio of about 0.10. The aerodynamic characteristics of this airfoil were then measured in the Langley 8-foot transonic pressure tunnel to evaluate experimentally the validity of the design technique.

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This report presents the results of this wind-tunnel investigation and compares them with the aerodynamic characteristics of an improved supercritical airfoil (supercritical airfoil 26a, ref. 5) which was experimentally developed and refined through extensive wind-tunnel testing with attention to both design and off-design conditions. Theoretical and experimental results of several other airfoils designed by use of the complex hodograph method of reference 4 are reported in references 6 and 7.

SYMBOLS

Values are given both in the International System of Units (SI) and in the U.S. Customary Units. Measurements and calculations are made in the U.S. Customary Units. On the computer prepared tables (tables II and III) C_p is designated CP and x/c is designated X/C.

C_p — pressure coefficient, $\dfrac{p_l - p_\infty}{q_\infty}$.

$C_{p,sonic}$ — pressure coefficient corresponding to local Mach number of 1.0

c — chord of airfoil, 63.5 cm (25.0 in.)

c_d — section drag coefficient, $\sum_{\text{Wake}} c_d' \dfrac{\Delta z}{c}$

c_d' — point drag coefficient (ref. 8)

$\Delta c_{d,s}$ — drag increment due to shock-wave losses

c_m — section pitching-moment coefficient about the quarter-chord point,

$$\sum_l C_p\left(0.25 - \frac{x}{c}\right)\frac{\Delta x}{c} - \sum_u C_p\left(0.25 - \frac{x}{c}\right)\frac{\Delta x}{c}$$

c_n — section normal-force coefficient, $\sum_l C_p \dfrac{\Delta x}{c} - \sum_u C_p \dfrac{\Delta x}{c}$

K — surface curvature, reciprocal of local radius of curvature

M — Mach number

m — surface slope, dy/dx

p static pressure, N/m^2 (lb/ft^2)

Δp_t total-pressure loss, N/m^2 (lb/ft^2)

q dynamic pressure, N/m^2 (lb/ft^2)

R Reynolds number based on airfoil chord

x ordinate along airfoil reference line measured from airfoil leading edge, cm (in.)

y ordinate normal to airfoil reference line, cm (in.)

z vertical distance in wake profile measured from bottom of rake, cm (in.)

α geometrical angle of attack of airfoil reference line, deg

Subscripts:

l local point on airfoil

max maximum

∞ undisturbed stream

Abbreviations:

l airfoil lower surface

u airfoil upper surface

AIRFOIL DESIGN APPROACHES

Airfoil sketches and surface slope and curvature distributions are shown in figures 1 to 3. Airfoil coordinates are presented in table I. Small surface irregularities are greatly exaggerated when examined from the standpoint of local curvature. Consequently, the curvature distributions presented in figure 3 were obtained by smoothing the airfoil coordinates and then fairing a curve through the curvature distribution calculated from these smoothed coordinates.

Experimentally Designed Airfoil

There seems to remain little doubt that for all practical purposes, a shock-free transonic flow can be realized experimentally. (See ref. 9.) During early phases of the NASA supercritical airfoil development program, an essentially complete elimination of the shock wave was achieved in a very narrow range of flow conditions. (See ref. 2.) It was later concluded, however, that off-design performance would suffer, particularly at normal-force coefficients below the design value, if undue emphasis was placed on a single-point shockless design goal.

The airfoil referred to herein as the experimental airfoil is an improved 10-percent-thick supercritical airfoil (supercritical airfoil 26a, ref. 5) which was developed and refined through extensive wind-tunnel testing with attention to both design and off-design conditions. This airfoil had good drag rise characteristics over a fairly wide range of normal-force coefficients from about 0.30 to 0.65 with no perceptible drag creep (gradual buildup of boundary layer and shock losses preceding drag divergence). The drag divergence Mach number varied from approximately 0.82 at $c_n = 0.30$ to 0.78 at $c_n = 0.80$.

Theoretically Designed Airfoil

Much of the material contained in this section is comprehensively discussed in reference 4 but repeated in abbreviated form herein for convenience. The detailed mathematics is beyond the intended scope of this report, however, and is not included.

The complex hodograph method.- The complex hodograph method of reference 4 is a technique for computing airfoil sections with shock-free transonic flow about them at a specific Mach number and angle of attack. It is an inverse solution whereby a smooth transonic potential flow is computed and then the body which generates the flow determined.

Briefly, the method is based on extending the physical plane and the hodograph plane into complex space where the problem of mapping from the complex hodograph space to the complex physical space is solved as a characteristic initial-value problem using a finite-difference scheme. With suitable selection of initial parameters, the method will generate shockless airfoil shapes (isentropic streamlines) in the real physical plane.

Viscous considerations.- The potential flow thus computed is supposed to be the inviscid flow outside the boundary layer of the actual airfoil. It has become increasingly apparent, however, that to arrive at any meaningful results in transonic flow problems, account must be taken of viscosity since the position and strength of the shock terminating a region of supersonic flow and the phenomenon of flow separation are strongly influenced by the presence of the boundary layer. (See, for example, ref. 10.) For this investigation the true geometric airfoil generating the computed flow was approximated by superimposing the effects of viscosity on the inviscid solution. This approximation was accomplished by subtracting from the calculated airfoil a displacement thickness obtained from a boundary-layer prediction scheme based on the method of Nash and MacDonald (ref. 11) using the calculated inviscid pressure distribution. Such a procedure is analogous to the development of the experimentally designed airfoil where the desired pressure distribution was developed in the wind tunnel on an effective airfoil shape which included both the geometrical airfoil and the boundary layer generated by the flow.

Design considerations.- In practice, some of the initial input to the design program may be determined by requiring the solution (airfoil shape) to satisfy certain requirements. The theoretical airfoil was required to be approximately 10 percent thick and to be shock free at a Mach number of 0.78 and a lift coefficient of about 0.6. These basic design guidelines were chosen because they were within the likely range of advanced technology aircraft applications, and because data on an experimentally designed airfoil with the same general design conditions were available for comparison.

Another design requirement was that after subtracting the boundary-layer displacement thickness, a thickness-chord ratio at the trailing edge of about 0.007 remains. This design consideration was established to provide a consistent basis for comparison with the experimental airfoil, which had a trailing-edge thickness-chord ratio of 0.007.

Resultant theoretical airfoil.- The airfoil shape before boundary-layer adjustment, theoretical design surface pressure distributions, and the shape of the supersonic zone are shown in figure 4. At design conditions, the supersonic region is large and extends over about 70 percent of the upper surface.

The boundary-layer scheme predicted separation near the 99-percent-chord station where the adverse pressure gradient at the trailing edge seemed to be overpredicted. Because of the overthickened boundary layer associated with this predicted separation region, irregularities appeared in the surface coordinates when the boundary-layer displacement thickness was subtracted from the design coordinates. Because such irregularities were not expected to occur physically, it was necessary to smooth the rear 7 or 8 percent of the upper surface.

To simplify model construction, the coordinate system of the theoretically designed airfoil was rotated so that it could be wrapped around an existing wind-tunnel model (supercritical airfoil 26a). These resultant coordinates, which include the boundary-layer displacement thickness adjustment and referred to hereafter as the theoretical airfoil, are the coordinates presented in table I.

Figures 1 and 2 show a remarkable similarity between the theoretical and experimental airfoils. The most significant dissimilarity is in the curvature distribution over the rear upper surface (fig. 3) which strongly influences off-design characteristics. Dissimilarities over the lower surfaces should not be significant since experience has indicated that supercritical type airfoils are not extremely sensitive to changes on the lower surface.

APPARATUS AND TECHNIQUES

Models

The wind-tunnel models, mounted in an inverted position, spanned the width of the tunnel with a span-chord ratio of 3.43. They were constructed with metal leading and trailing edges and with a metal core around which plastic fill was used to form the contours of the airfoils. Angle of attack was changed manually by rotating the model about pivots in the tunnel side walls. Sketches of one of the airfoils mounted in the tunnel and the profile drag rake are presented in figure 5 and a photograph of one of the airfoils and the profile drag rake mounted in the tunnel is shown as figure 6(a). Although not included in the sketches of figure 1, a trailing-edge cavity (fig. 6(b)) shown in reference 12 to have a favorable effect on the wake was included on both airfoils.

Wind Tunnel

The investigation was conducted in the Langley 8-foot transonic pressure tunnel. This tunnel is a continuous-flow, variable-pressure wind tunnel with controls that permit the independent variation of Mach number, stagnation pressure and temperature, and dewpoint. It has a 2.16-meter-square (85.2-inch-square) test section with filleted corners so that the total cross-sectional area is equivalent to that of a 2.44-meter-diameter (8-foot-diameter) circle. The upper and lower test-section walls are axially slotted to permit testing through the transonic speed range. The total slot width at the position of the model averaged about 5 percent of the width of the upper and lower walls.

The solid side walls and slotted upper and lower walls make this tunnel well suited to the investigation of two-dimensional models since the side walls act as end plates and the slots permit development of the flow field in the vertical direction.

Boundary-Layer Transition

Based on the technique discussed in reference 13, boundary-layer transition was fixed along the 28-percent-chord line on the upper and lower surfaces of the models in an attempt to simulate full-scale Reynolds numbers (fig. 7) by providing the same relative trailing-edge boundary-layer-displacement thickness at model scale as would exist at full-scale flight conditions. The simulation technique, which requires that laminar flow be maintained ahead of the transition trip, is limited on the upper surface to those test conditions in which shock waves or other steep adverse pressure gradients occur behind the point of fixed transition so that the flow is not tripped prematurely. Full-scale simulation on the lower surface would be valid through the Mach number range of the investigation since laminar flow can be maintained ahead of the trip for all conditions. The transition trips consisted of 0.25-cm-wide (0.10-inch) bands of number 90 carborundum grains.

In order that the experimental results for the theoretically designed and the experimentally designed airfoils be consistent, the theoretical boundary layer discussed previously was calculated at a Reynolds number of 20×10^6 with transition occurring between the 6- and 7-percent-chord line.

Measurements

Surface-pressure measurements.- Normal force and pitching moments acting on the airfoils were determined from surface static-pressure measurements. The surface-pressure measurements were obtained from a chordwise row of orifices located approximately 0.32c from the tunnel center line. Orifices were concentrated near the leading and trailing edges of the airfoil to define the severe pressure gradients in these regions. In addition, a rearward-facing orifice was included in the cavity at the trailing edge (identified at an upper surface x/c location of 1.00). Actual orifice locations are included in tables II and III. The transducers used in the differential pressure scanning valves to measure the static pressure at the airfoil surface had a range of ± 68.9 kN/m^2 (10 lb/in^2).

Wake measurements.- Drag forces acting on the airfoils, as measured by the momentum deficiency within the wake, were determined from vertical variations of the total and static pressures measured across the wake with the profile drag rake shown in figure 5(b). The rake was positioned in the vertical center-line plane of the tunnel, approximately 1 chord length rearward of the trailing edge of the airfoil. The total-pressure tubes were flattened horizontally and closely spaced vertically (0.36 percent of the airfoil chord) in the region of the wake associated with skin-friction boundary-layer losses. Outside this region, the tube vertical spacing progressively widened until in the region above the wing where only shock losses were anticipated, the total-pressure tubes were spaced apart about 7.2 percent of the chord. Static-pressure tubes were distributed as shown in figure 5(b). The rake was attached to the conventional center-line sting mount of the tunnel which permitted it to be moved vertically to center the close concentration of tubes in the boundary-layer wake.

Total and static pressures across the wake were also measured with the use of differential pressure scanning valves. The transducer in the valve connected to total-pressure tubes intended to measure boundary-layer losses had a range of ± 17.2 kN/m^2

(2.5 lb/in^2); and the transducer in the valve for measuring shock losses and static pressure had a range of ± 6.9 kN/m^2 (1 lb/in^2).

Reduction of Data

Calculation of c_n and c_m.- Section normal-force and pitching-moment coefficients were obtained by numerical integration (based on the trapezoidal method) of the local surface-pressure coefficient measured at each orifice multiplied by an appropriate weighting factor (incremental area).

Calculation of c_d.- To obtain section drag coefficients, point drag coefficients were computed for each total-pressure measurement in the wake by using the procedure of reference 8. These point drag coefficients were then summed by numerical integration across the wake (also based on the trapezoidal method).

Wind-Tunnel Wall Effects

Two main types of wind-tunnel-boundary interference effects which may be treated separately are solid and wake blockage at zero lift and lift-induced interference. Blockage effects are theoretically small for this particular model-tunnel configuration (for example, ref. 14); consequently, no corrections have been applied to the data to account for blockage effects. Lift interference manifests itself as an effective upward inclination (relative to the tunnel center line) of the stream approaching the inverted model. This flow angularity is proportional to the amount of lift generated by the model and results in the aerodynamic angle of attack being less than the measured geometric angle of attack, particularly at the higher lift coefficients. Experience has indicated, however, that the correction required to account for lift interference effect is generally much smaller than would be predicted by theory and because of this uncertainty, the uncorrected geometric angles of attack are used herein.

TEST CONDITIONS

Tests were conducted at Mach numbers from 0.50 to 0.83 for a stagnation pressure of 0.1013 MN/m^2 (1 atmosphere). The stagnation temperature of the tunnel air was automatically controlled at approximately 322 K (120° F) and the air was dried until the dewpoint in the test section was reduced sufficiently to avoid condensation effects. Resultant test Reynolds numbers based on the airfoil chord are as shown in figure 7.

PRESENTATION OF RESULTS

The data contained in this report are arranged in the following figures:

The wake profiles in figure 14 are representative of the momentum losses as indicated by stagnation pressure deficit across the wake. The middle section of these profiles reflect viscous and separation losses in the boundary layer, whereas, the "wings" of the profiles reflect direct losses in stagnation pressure across the shock waves. Drag increments due to shock-wave losses $\left(\Delta c_{d,s}\right)$ may be determined from integration of the drag measured across the wings. Drag divergence Mach number shown in figure 10 was defined as the point where the slope of the curve of section drag coefficient as a function of Mach number equaled 0.1.

DISCUSSION

Theoretical Airfoil

Basic aerodynamic characteristics.- Figures 9 and 10 indicate that near the design normal-force coefficient of 0.60, drag divergence occurs at a Mach number slightly higher than the design Mach number of 0.78. The drag increments shown in figure 12 also indicate that wave losses at $c_n = 0.6$ did not become significant until Mach numbers greater than about 0.79.

The theoretical airfoil experienced a gradual buildup of drag with increasing Mach number (referred to as drag creep) over the range of normal-force coefficients in figure 9. For c_n values of 0.55 and 0.60, the drag level dips near the design Mach number to a level below the subcritical ($M = 0.50$) drag level.

The increase in drag preceding the dip in the drag of the theoretical airfoil may be associated with the nature of the leading-edge recompression as illustrated in figures 11. Between $M = 0.50$ and 0.70, the increase was due to the influence of the recompression on the boundary layer since there are no perceptible wave losses at $c_n = 0.6$ in figure 12. As Mach number was increased to 0.74, wave losses began to appear.

Since wave losses appear in figure 12 to be about the same as $M = 0.74$ and at near-design Mach numbers, the dip in the drag must be accounted for by factors other than wave losses. Several interrelated factors are involved. First, and most important, the adverse interaction of the shock wave with the boundary layer would vary with Mach number. Figure 11 shows how the shock wave changes location and character as M increases from 0.74 to 0.78 and 0.79. Second, natural boundary-layer transition would be expected to occur ahead of the transition trip at the lower Mach number because of the pressure gradient through the forward recompression. As the Mach number increased above 0.74, the point of natural transition would move rearward to the transition trip ($x/c = 0.28$); thus, the extent of turbulent boundary-layer losses is reduced. Third, the lower drag level at near-design Mach numbers than at subcritical Mach numbers would be partially due to the reduced skin-friction losses at the higher test Reynolds number. (See fig. 7.)

Comparison of design and experimental pressure distributions.- A comparison between the computed pressure distribution and the nearest experimental pressure distribution is shown in figure 13. As with the Mach number discrepancy noted in the drag characteristics, the nearest experimental pressure distribution occurred at a Mach number ($M = 0.79$) about 0.01 higher than the design Mach number ($M = 0.78$).

A cursory analysis of this experimental pressure distribution would indicate that shockless flow was not achieved experimentally since a rather abrupt recompression is evident in the vicinity of the 60-percent-chord station. The wake profiles (fig. 14) and incremental drag attributed to wave losses (fig. 12) at these conditions, however, suggest a somewhat different interpretation. Although not truly shockless from a theoretical viewpoint, the airfoil approaches a minimum wave loss condition from an engineering viewpoint since only about four counts of wave drag $\left(\Delta c_{d,s} = 0.0004\right)$ may be associated with this pressure recompression. Profiles with such innocuous shock waves are in accordance with the concept of isentropically reducing the fieldwise extent of wave losses incorporated into supercritical airfoils. As Mach number was increased to 0.80, the shock wave moved rearward where it began to merge with the trailing-edge recovery and significant wave losses resulted. (See figs. 12 to 14.)

It is interesting to compare the local upstream Mach number entering the shock wave in the experimental pressure distributions of figure 13 with the generally accepted rule of thumb that with a normal shock wave, upstream Mach numbers of 1.15 can be tolerated without a drag rise. The local Mach number entering the shock wave (after some gradual recompression) of the $M = 0.79$ distribution (circle symbols) was approximately 1.1 whereas that of the $M = 0.80$ distribution (square symbols) had risen to almost 1.2 with a corresponding increase in wave losses.

The best drag-divergence characteristics ($M = 0.81$, fig. 10) seemed to occur at a normal-force coefficient of about 0.5, which is below the design value. Figure 12 shows that wave losses are practically zero at these conditions. The corresponding surface pressure distribution is shown in figure 24(d). These data suggest an empiricism that would involve specifying that the theoretically derived airfoil satisfy a slightly higher normal-force coefficient requirement than the desired design value.

Comparison With Experimental Airfoil

Aside from simply establishing the experimental characteristics of an airfoil theoretically designed for shockless transonic flow, it was the purpose of this investigation to see how these experimental characteristics compare with those of a supercritical airfoil experimentally designed for similar conditions.

<u>Normal-force and pitching-moment characteristics.</u>- Displacements in the normal-force and pitching-moment characteristics shown in figure 8 were due to the relatively lower aft camber of the theoretical airfoil.

<u>Drag characteristics.</u>- The subcritical drag levels (summarized in fig. 15 for $M = 0.50$) were generally lower for the theoretical airfoil. The pressure profiles of figure 16 indicate that the lower subcritical drag levels are due to the influence on the boundary layer of the lower leading-edge velocity peaks (less negative value of c_p) and the less adverse pressure gradient near the trailing edge. Such differences in the pressure profiles are, in turn, related to the minor physical differences between the two airfoils; the lower leading-edge velocity peak to the smaller leading-edge radius (table I), and the reduced adverse pressure gradient near the trailing edge to the lower aft camber. Up to normal-force coefficients near 0.6, the experimental airfoil did not evidence drag creep like that experienced by the theoretical airfoil (fig. 9) because of the more favorable recompression over the forward upper surface of the experimental airfoil.

Around the design normal-force coefficients (approximately $c_n = 0.5$ to 0.6) and at Mach numbers near those at which drag divergence occurs, the drag levels of the theoretical airfoil are a little lower than those of the experimental airfoil because of the previously discussed dip in the drag rise curves. However, there was good agreement in the drag divergence Mach numbers (fig. 10) at these, and higher, normal-force coefficients.

At normal-force coefficients less than 0.5, the drag divergence Mach numbers for the theoretical airfoil leveled off and were lower than those of the experimental airfoil. The poorer drag divergence characteristics at these low off-design conditions are caused by an overexpansion of the flow in the vicinity of the 70-percent chord (see, for example, figs. 24 and 25) with attendant boundary-layer and wave losses (fig. 12). The overexpansion was due to the relatively higher upper surface curvature (fig. 3) of the theoretical airfoil in this region.

<u>Necessity for off-design considerations.</u>- Although the aerodynamic characteristics of the two airfoils were generally very similar, subtle, but significant, differences were

present. These differences suggest that undue emphasis should not be placed on a single-point shockless design but that off-design behavior must be considered. The data also imply that in a practical situation, a more realistic design approach would be a trade-off between minimum wave losses at design conditions and acceptable characteristics at off-design conditions. Certainly, off-design characteristics cannot be ignored.

Evaluation of the Complex Hodograph Method

Overall, the experimental aerodynamic characteristics of the airfoil designed by the complex hodograph method were excellent. In addition, except for several off-design characteristics of the theoretically designed airfoil (drag creep and reduced drag divergence Mach numbers at low c_n) which were not as good as those of the experimentally designed supercritical airfoil, agreement between the two airfoils was generally good.

The results indicate that if used in conjunction with an adequate analysis program to evaluate off-design behavior, the complex hodograph method can be a valuable design tool.

CONCLUSIONS

The experimental aerodynamic characteristics of a 10-percent-thick airfoil theoretically designed to be shock free at Mach 0.78 and normal-force coefficient c_n of 0.59 have been evaluated and compared with those of an improved supercritical airfoil experimentally designed for similar design conditions. The results indicate the following major conclusions:

1. Except for slight degradation at off-design conditions (drag creep and reduced drag divergence Mach numbers at low c_n), the experimental aerodynamic characteristics of the theoretical airfoil compared well with those of the experimentally designed airfoil.

2. Undue emphasis on a single-point shockless design goal would more than likely compromise off-design characteristics. A more realistic design goal would be a minimum wave loss design point which would also provide acceptable off-design characteristics.

3. The complex hodograph design method can be a valuable design tool if used in conjunction with an adequate analysis program to evaluate off-design characteristics.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., May 21, 1974.

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TABLE I.- SECTION COORDINATES

[c = 63.5 cm (25 inches)]

x/c	Experimental airfoil		Theoretical airfoil	
	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$
0.0	0.0	0.0	0.0	0.0
.005	.0122	-.0128	.0111	-.0111
.01	.0163	-.0169	.0150	-.0151
.02	.0212	-.0217	.0200	-.0202
.03	.0244	-.0250	.0235	-.0240
.04	.0269	-.0276	.0262	-.0269
.05	.0290	-.0298	.0284	-.0294
.06	.0308	-.0317	.0302	-.0316
.07	.0324	-.0334	.0318	-.0334
.08	.0339	-.0349	.0333	-.0350
.09	.0352	-.0363	.0346	-.0364
.10	.0364	-.0376	.0358	-.0377
.11	.0375	-.0388	.0369	-.0389
.12	.0385	-.0399	.0380	-.0400
.13	.0395	-.0409	.0390	-.0409
.14	.0404	-.0418	.0399	-.0418
.15	.0412	-.0426	.0407	-.0426
.16	.0420	-.0434	.0415	-.0434
.17	.0427	-.0441	.0422	-.0441
.18	.0434	-.0448	.0429	-.0448
.19	.0440	-.0454	.0436	-.0454
.20	.0446	-.0460	.0442	-.0459
.21	.0452	-.0465	.0448	-.0464
.22	.0457	-.0470	.0454	-.0469
.23	.0462	-.0474	.0459	-.0473
.24	.0466	-.0478	.0464	-.0477
.25	.0470	-.0482	.0468	-.0481
.26	.0474	-.0485	.0472	-.0484
.27	.0477	-.0488	.0476	-.0487
0.28	0.0480	-0.0491	0.0480	-0.0490
.29	.0483	-.0493	.0483	-.0492
.30	.0486	-.0495	.0486	-.0494
.31	.0488	-.0497	.0489	-.0496
.32	.0490	-.0498	.0492	-.0497
.33	.0492	-.0449	.0494	-.0498
.34	.0494	-.0500	.0496	-.0499
.35	.0496	-.0500	.0498	-.0499
.36	.0497	-.0500	.0500	-.0499
.37	.0498	-.0500	.0502	-.0499
.38	.0499	-.0499	.0503	-.0498
.39	.0500	-.0498	.0504	-.0497
.40	.0500	-.0497	.0505	-.0496
.41	.0500	-.0495	.0506	-.0494
.42	.0500	-.0493	.0506	-.0492
.43	.0500	-.0491	.0506	-.0490
.44	.0500	-.0488	.0506	-.0488
.45	.0499	-.0485	.0506	-.0485
.46	.0498	-.0482	.0506	-.0482
.47	.0497	-.0478	.0505	-.0479
.48	.0496	-.0474	.0504	-.0475
.49	.0495	-.0469	.0503	-.0471
.50	.0493	-.0463	.0502	-.0467
.51	.0491	-.0457	.0501	-.0462
.52	.0489	-.0450	.0499	-.0457
.53	.0487	-.0443	.0497	-.0451
.54	.0485	-.0435	.0495	-.0445
.55	.0482	-.0427	.0493	-.0438
.56	.0479	-.0418	.0491	-.0431

TABLE I.- SECTION COORDINATES – Concluded

[c = 63.5 cm (25 inches)]

x/c	Experimental airfoil		Theoretical airfoil	
	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$
0.57	0.0476	-0.0408	0.0488	-0.0423
.58	.0473	-.0397	.0485	-.0415
.59	.0470	-.0386	.0482	-.0406
.60	.0466	-.0374	.0478	-.0397
.61	.0462	-.0361	.0474	-.0387
.62	.0458	-.0347	.0470	-.0377
.63	.0454	-.0332	.0466	-.0366
.64	.0450	-.0316	.0462	-.0354
.65	.0445	-.0299	.0457	-.0341
.66	.0440	-.0282	.0452	-.0328
.67	.0435	-.0264	.0446	-.0314
.68	.0430	-.0246	.0440	-.0299
.69	.0424	-.0227	.0434	-.0283
.70	.0418	-.0208	.0427	-.0267
.71	.0412	-.0189	.0420	-.0250
.72	.0406	-.0170	.0413	-.0232
.73	.0399	-.0151	.0405	-.0214
.74	.0392	-.0132	.0397	-.0195
.75	.0385	-.0114	.0389	-.0175
.76	.0377	-.0096	.0380	-.0155
.77	.0369	-.0078	.0371	-.0135
.78	.0361	-.0061	.0361	-.0115

x/c	Experimental airfoil		Theoretical airfoil	
	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$	$\left(\frac{y}{c}\right)_u$	$\left(\frac{y}{c}\right)_l$
0.79	0.0352	-0.0044	0.0351	-0.0095
.80	.0343	-.0028	.0341	-.0076
.81	.0333	-.0013	.0330	-.0058
.82	.0323	+.0001	.0319	-.0042
.83	.0312	.0014	.0308	-.0027
.84	.0301	.0026	.0296	-.0014
.85	.0289	.0036	.0284	-.0003
.86	.0277	.0045	.0272	+.0006
.87	.0264	.0052	.0259	.0014
.88	.0250	.0057	.0245	.0020
.89	.0235	.0060	.0230	.0024
.90	.0219	.0061	.0214	.0027
.91	.0202	.0061	.0197	.0028
.92	.0184	.0059	.0180	.0027
.93	.0165	.0054	.0162	.0024
.94	.0145	.0046	.0143	.0018
.95	.0124	.0035	.0123	+.0010
.96	.0102	.0021	.0102	.0
.97	.0079	+.0004	.0080	-.0011
.98	.0055	-.0016	.0056	-.0025
.99	.0029	-.0039	.0029	-.0044
1.00	-----	-.0066	-----	-.0069

TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a)

(a) $\alpha = -0.5^{o}$

	CP AT -											
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	1.066	1.094	1.126	1.143	1.147	1.155	1.162	1.166	1.166	1.174	1.177	0.000
.003	.464	.526	.622	.656	.681	.709	.723	.730	.751	.749	.770	.003
.012	-.385	-.319	-.232	-.157	-.145	-.060	-.068	-.070	-.038	-.014	.010	.012
.016	-.502	-.483	-.368	-.311	-.281	-.244	-.221	-.215	-.183	-.188	-.148	.016
.029	-.491	-.501	-.457	-.441	-.414	-.380	-.387	-.361	-.354	-.328	-.327	.029
.046	-.462	-.461	-.454	-.456	-.443	-.428	-.410	-.422	-.397	-.400	-.366	.046
.071	-.406	-.422	-.410	-.424	-.407	-.406	-.407	-.402	-.375	-.369	-.341	.071
.098	-.387	-.396	-.403	-.406	-.425	-.415	-.415	-.402	-.406	-.404	-.371	.098
.150	-.345	-.370	-.408	-.412	-.411	-.428	-.428	-.443	-.440	-.427	-.420	.150
.200	-.329	-.343	-.394	-.411	-.416	-.436	-.435	-.446	-.446	-.443	-.433	.200
.250	-.325	-.335	-.365	-.408	-.419	-.436	-.455	-.466	-.470	-.477	-.467	.250
.301	-.333	-.341	-.359	-.365	-.379	-.390	-.390	-.392	-.462	-.479	-.478	.301
.351	-.335	-.343	-.370	-.389	-.401	-.420	-.426	-.439	-.434	-.408	-.468	.351
.398	-.333	-.346	-.375	-.395	-.409	-.430	-.442	-.454	-.469	-.450	-.499	.398
.448	-.328	-.344	-.375	-.394	-.411	-.428	-.442	-.458	-.476	-.481	-.460	.448
.499	-.332	-.352	-.384	-.406	-.421	-.446	-.463	-.478	-.493	-.522	-.497	.499
.549	-.328	-.346	-.378	-.404	-.422	-.448	-.464	-.483	-.480	-.526	-.515	.549
.600	-.334	-.353	-.389	-.416	-.434	-.460	-.476	-.496	-.538	-.551	-.566	.600
.652	-.331	-.355	-.385	-.414	-.436	-.464	-.469	-.515	-.560	-.568	-.607	.652
.700	-.338	-.359	-.392	-.418	-.439	-.461	-.479	-.499	-.559	-.601	-.599	.700
.750	-.335	-.358	-.386	-.411	-.432	-.454	-.463	-.481	-.494	-.652	-.659	.750
.801	-.335	-.355	-.385	-.406	-.424	-.441	-.451	-.465	-.463	-.526	-.700	.801
.850	-.328	-.345	-.370	-.386	-.393	-.399	-.400	-.404	-.400	-.348	-.357	.850
.899	-.272	-.284	-.287	-.287	-.290	-.283	-.280	-.276	-.266	-.240	-.190	.899
.932	-.231	-.230	-.221	-.217	-.208	-.198	-.192	-.184	-.174	-.152	-.109	.932
.972	-.097	-.084	-.059	-.044	-.034	-.022	-.011	-.007	-.002	.011	.022	.972
.989	-.030	-.015	.014	.028	.037	.043	.049	.055	.059	.061	.064	.989
1.000	.019	.032	.053	.061	.064	.069	.076	.075	.075	.081	.077	1.000
LOWER SURFACE												
.006	.166	.174	.200	.215	.212	.212	.228	.240	.235	.245	.258	.006
.011	-.187	-.198	-.208	-.191	-.193	-.202	-.190	-.182	-.189	-.154	-.159	.011
.020	-.269	-.299	-.319	-.346	-.328	-.356	-.349	-.331	-.330	-.321	-.319	.020
.029	-.288	-.309	-.371	-.395	-.404	-.414	-.414	-.398	-.409	-.380	-.382	.029
.048	-.316	-.341	-.404	-.428	-.456	-.486	-.474	-.496	-.489	-.493	-.465	.048
.069	-.301	-.336	-.386	-.409	-.434	-.466	-.484	-.474	-.491	-.478	-.502	.069
.102	-.290	-.299	-.354	-.379	-.412	-.434	-.440	-.451	-.456	-.450	-.455	.102
.151	-.272	-.302	-.352	-.385	-.398	-.432	-.450	-.478	-.521	-.536	-.546	.151
.199	-.269	-.291	-.341	-.381	-.408	-.437	-.454	-.468	-.484	-.533	-.561	.199
.251	-.264	-.293	-.343	-.378	-.395	-.426	-.459	-.499	-.523	-.532	-.561	.251
.299	-.260	-.291	-.331	-.364	-.385	-.418	-.438	-.476	-.570	-.594	-.625	.299
.350	-.256	-.281	-.323	-.355	-.372	-.397	-.420	-.443	-.434	-.572	-.603	.350
.400	-.245	-.275	-.314	-.338	-.358	-.383	-.398	-.420	-.446	-.459	-.611	.400
.451	-.247	-.272	-.310	-.336	-.355	-.374	-.390	-.405	-.425	-.423	-.631	.451
.500	-.248	-.270	-.309	-.330	-.342	-.365	-.374	-.389	-.405	-.413	-.346	.500
.549	-.224	-.243	-.273	-.286	-.293	-.306	-.311	-.319	-.318	-.325	-.285	.549
.598	-.171	-.185	-.194	-.202	-.204	-.209	-.212	-.213	-.210	-.206	-.185	.598
.649	-.071	-.072	-.069	-.067	-.061	-.057	-.052	-.052	-.045	-.040	-.025	.649
.699	.070	.077	.095	.106	.110	.117	.118	.121	.125	.130	.136	.699
.747	.185	.197	.216	.227	.233	.239	.238	.241	.244	.246	.248	.747
.800	.286	.297	.319	.324	.327	.330	.332	.331	.334	.333	.335	.800
.850	.354	.364	.381	.389	.390	.395	.394	.390	.396	.392	.400	.850
.899	.386	.399	.418	.423	.431	.435	.432	.437	.443	.440	.439	.899
.929	.385	.400	.426	.434	.438	.442	.444	.444	.447	.444	.446	.929
.950	.378	.388	.414	.422	.429	.434	.435	.432	.444	.441	.440	.950
.998	.020	.038	.053	.066	.070	.077	.080	.080	.081	.082	.083	.998

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TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a) - Continued

(b) $\alpha = 0^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000					1.158	1.164	1.164	1.171	1.176	1.179	1.185	0.000
.003					.613	.633	.647	.669	.679	.693	.716	.003
.012					-.271	-.253	-.200	-.175	-.166	-.138	-.106	.012
.016					-.426	-.382	-.387	-.336	-.290	-.259	-.254	.016
.029					-.571	-.532	-.520	-.495	-.474	-.425	-.418	.029
.046					-.563	-.553	-.537	-.533	-.494	-.485	-.446	.046
.071					-.513	-.503	-.492	-.480	-.457	-.451	-.435	.071
.098					-.494	-.507	-.494	-.477	-.471	-.461	-.451	.098
.150					-.486	-.504	-.504	-.515	-.519	-.507	-.490	.150
.200					-.463	-.497	-.506	-.505	-.498	-.510	-.487	.200
.250					-.467	-.502	-.519	-.526	-.527	-.531	-.519	.250
.301					-.413	-.431	-.453	-.507	-.534	-.528	-.519	.301
.351					-.439	-.458	-.465	-.456	-.507	-.529	-.521	.351
.398					-.441	-.463	-.481	-.493	-.435	-.536	-.547	.398
.448					-.444	-.467	-.485	-.499	-.482	-.539	-.538	.448
.499					-.450	-.476	-.493	-.498	-.535	-.527	-.578	.499
.549					-.445	-.473	-.496	-.508	-.543	-.537	-.545	.549
.600					-.460	-.477	-.508	-.543	-.558	-.586	-.586	.600
.652					-.455	-.482	-.512	-.540	-.577	-.621	-.624	.652
.700					-.452	-.480	-.497	-.532	-.602	-.606	-.672	.700
.750					-.443	-.468	-.478	-.488	-.540	-.660	-.722	.750
.801					-.438	-.456	-.464	-.473	-.448	-.569	-.745	.801
.850					-.404	-.410	-.410	-.407	-.393	-.345	-.349	.850
.899					-.293	-.289	-.285	-.276	-.260	-.228	-.186	.899
.932					-.216	-.208	-.196	-.184	-.170	-.148	-.112	.932
.972					-.038	-.028	-.018	-.011	-.001	.009	.017	.972
.989					.032	.040	.042	.047	.054	.059	.056	.989
1.000					.060	.062	.064	.071	.074	.072	.067	1.000
LOWER SURFACE												
.006					.315	.324	.327	.332	.325	.331	.330	.006
.011					-.083	-.076	-.072	-.086	-.066	-.063	-.081	.011
.020					-.232	-.211	-.239	-.239	-.221	-.222	-.229	.020
.029					-.284	-.294	-.274	-.295	-.315	-.315	-.297	.029
.048					-.338	-.355	-.358	-.370	-.371	-.403	-.386	.048
.069					-.346	-.358	-.375	-.375	-.399	-.404	-.415	.069
.102					-.327	-.352	-.361	-.375	-.394	-.393	-.388	.102
.151					-.344	-.360	-.380	-.408	-.417	-.440	-.465	.151
.199					-.345	-.375	-.396	-.409	-.418	-.440	-.446	.199
.251					-.345	-.381	-.399	-.416	-.444	-.486	-.481	.251
.299					-.341	-.366	-.382	-.391	-.432	-.526	-.570	.299
.350					-.336	-.362	-.377	-.387	-.410	-.408	-.544	.350
.400					-.325	-.352	-.365	-.373	-.405	-.434	-.513	.400
.451					-.324	-.349	-.366	-.374	-.395	-.418	-.410	.451
.500					-.327	-.347	-.358	-.361	-.378	-.403	-.414	.500
.549					-.274	-.287	-.297	-.299	-.308	-.313	-.313	.549
.598					-.196	-.199	-.200	-.202	-.203	-.201	-.198	.598
.649					-.055	-.048	-.049	-.044	-.039	-.037	-.033	.649
.699					.117	.120	.123	.130	.134	.135	.139	.699
.747					.242	.245	.248	.254	.255	.257	.259	.747
.800					.343	.345	.345	.352	.351	.350	.354	.800
.850					.409	.408	.412	.413	.414	.412	.417	.850
.899					.447	.449	.452	.454	.453	.453	.457	.899
.929					.450	.457	.455	.459	.461	.458	.462	.929
.950					.437	.442	.450	.451	.456	.452	.454	.950
.998					.064	.068	.072	.077	.079	.077	.075	.998

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TABLE II.- SURFACE PRESSURE DISTRIBUTIONS: EXPERIMENTAL AIRFOIL (26a) - Continued

(c) $\alpha = 0.5^{\circ}$

					CP AT -							
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
					UPPER SURFACE							
0.000					1.155	1.159	1.169	1.171	1.177	1.183	1.187	0.000
.003					.502	.546	.556	.588	.603	.627	.648	.003
.012					-.425	-.405	-.341	-.324	-.296	-.258	-.220	.012
.016					-.607	-.546	-.493	-.479	-.418	-.423	-.365	.016
.029					-.757	-.706	-.655	-.652	-.601	-.557	-.529	.029
.046					-.737	-.724	-.719	-.691	-.689	-.610	-.586	.046
.071					-.636	-.696	-.643	-.643	-.584	-.555	-.540	.071
.098					-.608	-.627	-.608	-.590	-.570	-.594	-.521	.098
.150					-.573	-.620	-.628	-.608	-.590	-.563	-.552	.150
.200					-.521	-.564	-.637	-.626	-.621	-.604	-.569	.200
.250					-.499	-.553	-.605	-.615	-.623	-.612	-.592	.250
.301					-.495	-.549	-.585	-.605	-.615	-.609	-.589	.301
.351					-.497	-.511	-.487	-.582	-.598	-.600	-.602	.351
.398					-.488	-.520	-.524	-.583	-.631	-.628	-.622	.398
.448					-.482	-.510	-.534	-.483	-.620	-.626	-.621	.448
.499					-.487	-.519	-.537	-.532	-.624	-.639	-.638	.499
.549					-.475	-.511	-.526	-.553	-.517	-.660	-.653	.549
.600					-.477	-.509	-.534	-.568	-.589	-.698	-.697	.600
.652					-.476	-.515	-.541	-.595	-.628	-.655	-.715	.652
.700					-.472	-.494	-.509	-.580	-.557	-.679	-.699	.700
.750					-.459	-.476	-.493	-.464	-.517	-.753	-.757	.750
.801					-.437	-.461	-.468	-.465	-.452	-.488	-.782	.801
.850					-.405	-.405	-.405	-.399	-.394	-.304	-.324	.850
.899					-.290	-.285	-.283	-.270	-.258	-.207	-.164	.899
.932					-.206	-.196	-.188	-.174	-.172	-.129	-.091	.932
.972					-.030	-.025	-.011	-.011	-.003	.011	.007	.972
.989					.027	.038	.047	.047	.053	.055	.037	.989
1.000					.054	.060	.061	.066	.066	.068	.046	1.000
					LOWER SURFACE							
.006					.428	.446	.452	.452	.448	.424	.425	.006
.011					.064	.052	.066	.048	.052	.047	.035	.011
.020					-.097	-.066	-.098	-.084	-.087	-.092	-.103	.020
.029					-.145	-.150	-.168	-.155	-.187	-.150	-.179	.029
.048					-.216	-.230	-.253	-.235	-.258	-.272	-.270	.048
.069					-.224	-.242	-.237	-.248	-.238	-.274	-.282	.069
.102					-.236	-.240	-.255	-.264	-.289	-.300	-.307	.102
.151					-.260	-.274	-.286	-.307	-.318	-.315	-.359	.151
.199					-.278	-.291	-.304	-.318	-.332	-.353	-.367	.199
.251					-.286	-.306	-.312	-.331	-.351	-.379	-.417	.251
.299					-.291	-.308	-.319	-.333	-.348	-.374	-.426	.299
.350					-.291	-.309	-.320	-.330	-.349	-.365	-.389	.350
.400					-.281	-.307	-.316	-.325	-.340	-.362	-.404	.400
.451					-.287	-.308	-.316	-.333	-.351	-.365	-.403	.451
.500					-.287	-.306	-.319	-.328	-.342	-.357	-.393	.500
.549					-.252	-.259	-.267	-.274	-.281	-.291	-.305	.549
.598					-.173	-.179	-.178	-.183	-.178	-.182	-.190	.598
.649					-.034	-.034	-.031	-.034	-.032	-.025	-.029	.649
.699					.124	.131	.135	.141	.145	.146	.143	.699
.747					.248	.255	.260	.260	.264	.268	.267	.747
.800					.347	.350	.352	.355	.354	.360	.358	.800
.850					.414	.412	.417	.421	.416	.422	.422	.850
.899					.445	.450	.456	.458	.454	.459	.459	.899
.929					.450	.454	.456	.463	.466	.464	.468	.929
.950					.443	.448	.451	.457	.455	.460	.457	.950
.998					.059	.066	.068	.072	.077	.079	.046	.998

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TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a) - Continued

(d) $\alpha = 1.0^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	1.019	1.063	1.110	1.137	1.150	1.159	1.163	1.167	1.171	1.181	1.181	0.000
.003	-.022	[illegible]	.259	.376	.408	.452	.481	.506	.536	.559	.580	.003
.012	-1.043	[illegible]	-.777	-.657	-.586	-.479	-.448	-.428	-.389	-.331	-.294	.012
.016	-1.029	-1.056	-.954	-.849	-.728	-.671	-.625	-.575	-.532	-.484	-.433	.016
.029	-.895	-.958	-1.012	-.959	-.903	-.810	-.794	-.750	-.717	-.664	-.618	.029
.046	-.783	-.790	-.866	-.937	-.950	-.883	-.858	-.808	-.768	-.726	-.673	.046
.071	-.636	-.658	-.721	-.846	-.853	-.831	-.785	-.749	-.743	-.675	-.647	.071
.098	-.561	-.586	-.625	-.634	-.738	-.844	-.790	-.756	-.741	-.695	-.659	.098
.150	-.497	-.525	-.570	-.608	-.608	-.719	-.730	-.705	-.693	-.652	-.628	.150
.200	-.463	-.490	-.540	-.572	-.595	-.682	-.697	-.683	-.679	-.653	-.627	.200
.250	-.446	-.460	-.509	-.550	-.573	-.651	-.692	-.699	-.698	-.677	-.638	.250
.301	-.434	-.454	-.500	-.537	-.577	-.589	-.691	-.695	-.690	-.663	-.643	.301
.351	-.422	-.439	-.488	-.516	-.537	-.518	-.653	-.698	-.689	-.676	-.652	.351
.398	-.413	-.430	-.475	-.501	-.523	-.543	-.665	-.692	-.702	-.685	-.666	.396
.448	-.400	-.420	-.463	-.491	-.512	-.547	-.492	-.704	-.706	-.695	-.681	.448
.499	-.397	-.417	-.462	-.491	-.511	-.551	-.517	-.703	-.703	-.712	-.695	.499
.549	-.384	-.405	-.444	-.483	-.499	-.538	-.542	-.694	-.715	-.719	-.699	.549
.600	-.384	-.409	-.448	-.480	-.493	-.544	-.592	-.534	-.756	-.755	-.746	.600
.652	-.377	-.402	-.441	-.465	-.488	-.531	-.553	-.477	-.764	-.790	-.780	.652
.700	-.378	-.395	-.436	-.466	-.479	-.520	-.535	-.494	-.590	-.821	-.809	.700
.750	-.367	-.386	-.423	-.443	-.462	-.480	-.491	-.490	-.406	-.862	-.861	.750
.801	-.362	-.379	-.410	-.427	-.436	-.458	-.466	-.475	-.393	-.522	-.793	.801
.850	-.344	-.363	-.378	-.390	-.400	-.405	-.403	-.396	-.360	-.279	-.297	.850
.899	-.286	-.288	-.292	-.288	-.278	-.279	-.281	-.266	-.238	-.172	-.171	.899
.932	-.233	-.228	-.220	-.209	-.197	-.187	-.180	-.178	-.158	-.103	-.105	.932
.972	-.093	-.080	-.057	-.039	-.030	-.018	-.012	-.007	-.000	.010	-.024	.972
.989	-.032	-.015	.009	.019	.029	.036	.044	.048	.052	.043	.000	.989
1.000	.011	.021	.035	.043	.047	.062	.061	.062	.067	.056	.019	1.000
LOWER SURFACE												
.006	.600	.570	.580	.566	.552	.553	.553	.540	.536	.524	.496	.006
.011	.229	.237	.220	.183	.196	.184	.180	.177	.152	.146	.133	.011
.020	.106	.085	.060	.057	.051	.014	.019	.030	.019	-.013	-.028	.020
.029	.011	.010	-.009	-.011	-.050	-.040	-.051	-.064	-.074	-.088	-.110	.029
.048	-.055	-.065	-.094	-.114	-.120	-.147	-.133	-.147	-.151	-.170	-.182	.048
.069	-.076	-.115	-.128	-.154	-.150	-.159	-.156	-.177	-.183	-.196	-.215	.069
.102	-.104	-.131	-.150	-.155	-.162	-.176	-.184	-.191	-.192	-.216	-.230	.102
.151	-.137	-.155	-.176	-.186	-.205	-.199	-.218	-.227	-.229	-.251	-.278	.151
.199	-.149	-.173	-.192	-.210	-.219	-.233	-.243	-.256	-.260	-.282	-.304	.199
.251	-.168	-.184	-.207	-.221	-.229	-.247	-.257	-.266	-.279	-.305	-.341	.251
.299	-.148	-.168	-.205	-.221	-.235	-.248	-.262	-.270	-.279	-.306	-.344	.299
.350	-.163	-.177	-.210	-.231	-.240	-.254	-.266	-.281	-.287	-.314	-.350	.350
.400	-.165	-.184	-.215	-.234	-.248	-.260	-.272	-.284	-.292	-.313	-.360	.400
.451	-.180	-.197	-.225	-.244	-.252	-.272	-.280	-.292	-.293	-.316	-.364	.451
.500	-.187	-.200	-.230	-.249	-.261	-.274	-.283	-.291	-.302	-.323	-.368	.500
.549	-.167	-.186	-.210	-.219	-.225	-.237	-.239	-.245	-.245	-.261	-.298	.549
.598	-.126	-.136	-.145	-.156	-.158	-.162	-.160	-.160	-.161	-.170	-.186	.598
.649	-.033	-.038	-.030	-.029	-.024	-.024	-.027	-.021	-.021	-.020	-.033	.649
.699	.096	.104	.119	.132	.133	.143	.148	.149	.151	.150	.145	.699
.747	.208	.226	.243	.250	.259	.268	.271	.275	.276	.274	.266	.747
.800	.308	.323	.345	.354	.357	.366	.371	.375	.373	.372	.361	.800
.850	.375	.392	.411	.420	.425	.433	.433	.436	.438	.438	.430	.850
.899	.409	.425	.449	.457	.459	.468	.471	.476	.475	.473	.464	.899
.929	.404	.423	.448	.457	.464	.470	.472	.477	.479	.478	.465	.929
.950	.393	.412	.430	.445	.453	.463	.461	.466	.469	.466	.454	.950
.998	.012	.021	.033	.046	.046	.057	.061	.067	.070	.056	.017	.998

635 1

TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a) - Continued

(e) $\alpha = 1.5^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.981	1.030	1.094	1.121	1.131	1.146	1.152	1.157	1.164	1.182		0.000
.003	-.259	-.110	.158	.254	.312	.383	.396	.449	.464	.503		.003
.012	-1.293	-1.269	-.969	-.790	-.696	-.605	-.567	-.514	-.467	-.413		.012
.016	-1.310	-1.351	-1.147	-.976	-.864	-.777	-.731	-.694	-.644	-.600		.016
.029	-1.048	-1.108	-1.264	-1.112	-.999	-.911	-.891	-.847	-.789	-.747		.029
.046	-.896	-.940	-1.142	-1.151	-1.087	-.981	-.962	-.907	-.866	-.798		.046
.071	-.696	-.714	-.958	-1.096	-1.031	-.945	-.913	-.870	-.823	-.785		.071
.098	-.637	-.665	-.684	-.989	-.994	-.936	-.914	-.863	-.822	-.794		.098
.150	-.559	-.587	-.648	-.506	-.894	-.855	-.844	-.821	-.794	-.741		.150
.200	-.508	-.533	-.595	-.607	-.842	-.861	-.839	-.817	-.781	-.738		.200
.250	-.480	-.508	-.555	-.586	-.459	-.847	-.843	-.819	-.774	-.749		.250
.301	-.460	-.487	-.539	-.577	-.568	-.811	-.802	-.775	-.762	-.738		.301
.351	-.453	-.470	-.519	-.545	-.558	-.780	-.774	-.779	-.754	-.731		.351
.398	-.435	-.458	-.507	-.536	-.557	-.523	-.792	-.795	-.782	-.751		.398
.448	-.424	-.449	-.490	-.527	-.547	-.428	-.792	-.800	-.793	-.763		.448
.499	-.414	-.441	-.482	-.518	-.540	-.484	-.775	-.792	-.780	-.763		.499
.549	-.406	-.430	-.465	-.499	-.530	-.497	-.575	-.792	-.784	-.770		.549
.600	-.399	-.421	-.472	-.494	-.520	-.534	-.417	-.825	-.825	-.807		.600
.652	-.392	-.414	-.455	-.487	-.510	-.536	-.448	-.845	-.854	-.842		.652
.700	-.383	-.412	-.447	-.478	-.499	-.519	-.497	-.662	-.880	-.872		.700
.750	-.374	-.398	-.434	-.458	-.475	-.483	-.483	-.356	-.895	-.922		.750
.801	-.362	-.390	-.413	-.438	-.446	-.467	-.462	-.385	-.367	-.512		.801
.850	-.350	-.371	-.385	-.390	-.403	-.407	-.401	-.350	-.263	-.272		.850
.899	-.282	-.290	-.290	-.282	-.289	-.284	-.275	-.250	-.177	-.163		.899
.932	-.229	-.231	-.211	-.203	-.199	-.190	-.187	-.166	-.113	-.099		.932
.972	-.096	-.080	-.054	-.035	-.035	-.018	-.016	-.008	.007	-.020		.972
.989	-.031	-.018	.009	.019	.024	.037	.045	.047	.043	-.004		.989
1.000	.008	.021	.034	.044	.049	.058	.065	.067	.059	.016		1.000
LOWER SURFACE												
.006	.699	.690	.685	.663	.659	.649	.641	.637	.620	.601		.006
.011	.375	.374	.336	.329	.307	.287	.295	.256	.257	.240		.011
.020	.209	.203	.191	.165	.147	.146	.137	.125	.104	.087		.020
.029	.133	.118	.096	.082	.054	.057	.047	.043	.033	.018		.029
.048	.025	.010	-.003	-.016	-.050	-.049	-.042	-.052	-.068	-.081		.048
.069	-.021	-.032	-.046	-.061	-.061	-.082	-.077	-.092	-.095	-.128		.069
.102	-.056	-.073	-.089	-.096	-.106	-.115	-.117	-.128	-.140	-.152		.102
.151	-.084	-.105	-.120	-.134	-.148	-.147	-.145	-.165	-.178	-.193		.151
.199	-.106	-.120	-.137	-.155	-.173	-.183	-.188	-.191	-.201	-.228		.199
.251	-.127	-.139	-.165	-.182	-.189	-.198	-.203	-.214	-.227	-.260		.251
.299	-.119	-.126	-.154	-.167	-.180	-.187	-.198	-.203	-.225	-.237		.299
.350	-.138	-.149	-.175	-.190	-.200	-.213	-.220	-.229	-.244	-.265		.350
.400	-.138	-.151	-.179	-.187	-.204	-.218	-.224	-.235	-.242	-.277		.400
.451	-.160	-.167	-.199	-.216	-.222	-.229	-.244	-.248	-.262	-.286		.451
.500	-.165	-.184	-.207	-.225	-.233	-.244	-.242	-.254	-.267	-.297		.500
.549	-.151	-.164	-.181	-.195	-.200	-.204	-.216	-.217	-.232	-.249		.549
.598	-.112	-.120	-.123	-.134	-.141	-.142	-.138	-.140	-.145	-.163		.598
.649	-.026	-.018	-.022	-.015	-.015	-.006	-.004	-.003	-.008	-.015		.649
.699	.105	.117	.134	.141	.142	.153	.158	.160	.159	.156		.699
.747	.221	.229	.254	.264	.267	.273	.281	.285	.285	.276		.747
.800	.314	.332	.352	.368	.370	.378	.380	.383	.386	.380		.800
.850	.381	.401	.424	.435	.439	.439	.447	.448	.452	.448		.850
.899	.406	.434	.458	.467	.471	.477	.485	.489	.487	.482		.899
.929	.410	.433	.453	.467	.474	.479	.485	.487	.490	.480		.929
.950	.394	.423	.445	.453	.464	.471	.471	.473	.476	.471		.950
.998	.011	.018	.034	.040	.045	.057	.067	.071	.059	.027		.998

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TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a) - Continued

(f) $\alpha = 2.0^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.897	.974	1.061	1.096	1.115	1.130	1.141	1.155	1.162	1.173		0.000
.003	-.471	-.246	.027	.156	.233	.307	.331	.369	.407	.458		.003
.012	-1.551	-1.473	-1.089	-.884	-.779	-.702	-.640	-.570	-.525	-.447		.012
.016	-1.520	-1.679	-1.283	-1.061	-.958	-.861	-.813	-.759	-.709	-.633		.016
.029	-1.212	-1.344	-1.404	-1.226	-1.101	-1.025	-.955	-.898	-.848	-.799		.029
.046	-.939	-1.011	-1.426	-1.237	-1.177	-1.070	-1.002	-.962	-.924	-.851		.046
.071	-.800	-.846	-1.258	-1.215	-1.129	-1.022	-.984	-.945	-.882	-.846		.071
.098	-.700	-.742	-.793	-1.172	-1.109	-1.031	-.987	-.935	-.884	-.862		.098
.150	-.604	-.649	-.662	-1.097	-1.029	-.971	-.947	-.902	-.837	-.798		.150
.200	-.541	-.590	-.630	-.890	-.998	-.936	-.931	-.870	-.852	-.790		.200
.250	-.514	-.548	-.595	-.480	-.938	-.927	-.909	-.872	-.848	-.803		.250
.301	-.491	-.525	-.576	-.556	-.927	-.904	-.873	-.848	-.820	-.780		.301
.351	-.474	-.503	-.547	-.559	-.458	-.893	-.888	-.868	-.824	-.777		.351
.398	-.457	-.483	-.526	-.551	-.450	-.907	-.890	-.852	-.844	-.801		.398
.448	-.445	-.471	-.512	-.536	-.493	-.902	-.885	-.862	-.837	-.808		.448
.499	-.432	-.462	-.504	-.531	-.530	-.881	-.894	-.861	-.853	-.827		.499
.549	-.416	-.444	-.490	-.518	-.520	-.484	-.875	-.855	-.842	-.803		.549
.600	-.416	-.449	-.486	-.520	-.524	-.413	-.863	-.896	-.878	-.852		.600
.652	-.407	-.429	-.469	-.497	-.509	-.435	-.559	-.918	-.904	-.874		.652
.700	-.396	-.421	-.464	-.481	-.503	-.460	-.373	-.942	-.926	-.903		.700
.750	-.386	-.409	-.436	-.461	-.475	-.448	-.364	-.511	-.957	-.950		.750
.801	-.373	-.391	-.412	-.445	-.458	-.438	-.368	-.328	-.395	-.410		.801
.850	-.353	-.369	-.384	-.399	-.401	-.384	-.327	-.256	-.256	-.273		.850
.899	-.282	-.288	-.288	-.289	-.285	-.267	-.233	-.171	-.151	-.193		.899
.932	-.231	-.226	-.208	-.198	-.200	-.183	-.156	-.101	-.093	-.145		.932
.972	-.090	-.080	-.054	-.039	-.031	-.018	-.009	.006	-.022	-.083		.972
.989	-.026	-.017	.006	.020	.029	.040	.047	.045	-.004	-.055		.989
1.000	.008	.018	.029	.041	.048	.059	.062	.054	.007	-.041		1.000
LOWER SURFACE												
.006	.805	.780	.756	.747	.746	.715	.721	.685	.680	.669		.006
.011	.469	.444	.432	.422	.394	.364	.371	.355	.324	.304		.011
.020	.308	.275	.265	.258	.233	.235	.221	.188	.175	.136		.020
.029	.207	.204	.178	.162	.147	.132	.140	.116	.103	.070		.029
.048	.094	.092	.068	.061	.033	.041	.027	.007	-.007	-.021		.048
.069	.038	.040	.020	.006	.008	-.010	-.024	-.031	-.032	-.073		.069
.102	-.001	-.011	-.033	-.035	-.043	-.062	-.059	-.061	-.088	-.093		.102
.151	-.042	-.057	-.066	-.088	-.086	-.093	-.093	-.110	-.126	-.127		.151
.199	-.079	-.085	-.096	-.107	-.116	-.127	-.126	-.136	-.160	-.187		.199
.251	-.098	-.105	-.131	-.138	-.144	-.157	-.153	-.169	-.181	-.217		.251
.299	-.090	-.098	-.118	-.123	-.143	-.152	-.154	-.172	-.198	-.233		.299
.350	-.111	-.124	-.146	-.157	-.163	-.170	-.181	-.189	-.215	-.248		.350
.400	-.118	-.130	-.153	-.167	-.172	-.184	-.183	-.195	-.219	-.256		.400
.451	-.135	-.151	-.168	-.183	-.192	-.190	-.202	-.212	-.240	-.264		.451
.500	-.153	-.163	-.180	-.194	-.204	-.208	-.218	-.219	-.253	-.281		.500
.549	-.133	-.145	-.163	-.169	-.176	-.176	-.181	-.190	-.211	-.238		.549
.598	-.096	-.104	-.113	-.118	-.115	-.117	-.119	-.133	-.135	-.160		.598
.649	-.013	-.020	-.006	-.007	.001	.006	.013	.005	.001	-.017		.649
.699	.114	.124	.138	.150	.157	.165	.167	.170	.164	.142		.699
.747	.225	.240	.251	.276	.277	.285	.290	.289	.279	.270		.747
.800	.318	.339	.360	.373	.379	.385	.389	.393	.380	.364		.800
.850	.383	.403	.429	.438	.444	.450	.452	.453	.445	.432		.850
.899	.415	.434	.458	.469	.477	.489	.489	.489	.477	.462		.899
.929	.415	.430	.459	.469	.479	.488	.492	.493	.478	.461		.929
.950	.397	.418	.448	.459	.466	.476	.479	.482	.462	.448		.950
.998	.009	.017	.026	.039	.042	.061	.062	.050	.018	-.064		.998

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TABLE II.- SURFACE PRESSURE DISTRIBUTIONS: EXPERIMENTAL AIRFOIL (26a) - Continued

(g) $\alpha = 2.5^{\circ}$

	CP AT -										
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
					UPPER SURFACE							
0.000	.844	.924	1.029	1.075	1.094	1.121	1.135	1.137				0.000
.003	-.691	-.430	-.057	.064	.153	.231	.265	.318				.003
.012	-1.827	-1.744	-1.184	-.973	-.660	-.785	-.721	-.663				.012
.016	-1.788	-1.966	-1.380	-1.160	-1.058	-.950	-.893	-.826				.016
.029	-1.430	-1.753	-1.522	-1.319	-1.217	-1.094	-1.050	-.988				.029
.046	-1.097	-1.057	-1.572	-1.358	-1.254	-1.144	-1.094	-1.044				.046
.071	-.893	-.943	-1.497	-1.327	-1.230	-1.125	-1.076	-1.018				.071
.098	-.783	-.821	-1.383	-1.269	-1.186	-1.101	-1.054	-1.000				.098
.150	-.665	-.706	-.514	-1.195	-1.134	-1.068	-1.035	-.966				.150
.200	-.597	-.635	-.593	-1.159	-1.106	-1.030	-.999	-.959				.200
.250	-.551	-.588	-.600	-1.131	-1.103	-1.039	-1.008	-.970				.250
.301	-.526	-.557	-.595	-.863	-1.087	-1.029	-.994	-.953				.301
.351	-.509	-.531	-.577	-.424	-1.041	-.984	-.959	-.916				.351
.398	-.485	-.509	-.559	-.472	-1.026	-1.002	-.962	-.918				.398
.448	-.469	-.492	-.541	-.497	-.686	-.979	-.942	-.922				.448
.499	-.460	-.482	-.525	-.512	-.369	-.985	-.948	-.936				.499
.549	-.433	-.462	-.505	-.512	-.410	-.979	-.976	-.942				.549
.600	-.430	-.457	-.502	-.513	-.456	-.995	-.989	-.967				.600
.652	-.421	-.438	-.482	-.499	-.472	-.875	-1.005	-.985				.652
.700	-.409	-.433	-.464	-.490	-.477	-.408	-.996	-1.000				.700
.750	-.395	-.417	-.446	-.471	-.465	-.340	-.439	-.933				.750
.801	-.377	-.401	-.424	-.454	-.435	-.343	-.311	-.388				.801
.850	-.355	-.374	-.380	-.397	-.400	-.328	-.263	-.266				.850
.899	-.284	-.290	-.280	-.292	-.286	-.245	-.188	-.166				.899
.932	-.234	-.219	-.210	-.208	-.199	-.164	-.123	-.098				.932
.972	-.086	-.076	-.053	-.034	-.029	-.009	.004	-.006				.972
.989	-.026	-.011	.001	.027	.035	.047	.048	.030				.989
1.000	.006	.019	.032	.049	.062	.073	.069	.024				1.000
					LOWER SURFACE							
.006	.864	.864	.827	.817	.797	.800	.766	.764				.006
.011	.570	.559	.514	.524	.488	.469	.462	.420				.011
.020	.389	.394	.348	.347	.330	.325	.302	.279				.020
.029	.291	.301	.255	.253	.244	.230	.209	.187				.029
.048	.177	.175	.135	.137	.134	.112	.104	.085				.048
.069	.123	.110	.105	.092	.060	.065	.060	.042				.069
.102	.053	.051	.033	.036	.030	.017	.004	-.010				.102
.151	.005	-.003	-.020	-.024	-.028	-.035	-.036	-.056				.151
.199	-.030	-.043	-.059	-.068	-.064	-.073	-.073	-.094				.199
.251	-.059	-.069	-.086	-.089	-.091	-.104	-.113	-.120				.251
.299	-.058	-.064	-.082	-.094	-.098	-.100	-.110	-.123				.299
.350	-.079	-.095	-.110	-.117	-.119	-.131	-.137	-.156				.350
.400	-.088	-.098	-.123	-.128	-.130	-.138	-.149	-.172				.400
.451	-.112	-.120	-.143	-.147	-.147	-.156	-.164	-.190				.451
.500	-.126	-.133	-.158	-.163	-.166	-.175	-.189	-.209				.500
.549	-.113	-.123	-.139	-.138	-.141	-.153	-.151	-.166				.549
.598	-.076	-.082	-.092	-.088	-.088	-.095	-.092	-.104				.598
.649	.002	.005	.005	.020	.019	.025	.028	.016				.649
.699	.125	.139	.150	.169	.173	.179	.184	.179				.699
.747	.231	.247	.268	.285	.287	.299	.300	.292				.747
.800	.328	.346	.370	.380	.387	.394	.402	.396				.800
.850	.391	.408	.431	.441	.453	.462	.465	.457				.850
.899	.420	.439	.468	.477	.483	.495	.499	.495				.899
.929	.418	.437	.463	.472	.478	.491	.498	.490				.929
.950	.399	.428	.448	.456	.468	.483	.485	.478				.950
.998	.011	.014	.026	.051	.071	.075	.078	.035				.998

635 5

TABLE II.- SURFACE PRESSURE DISTRIBUTIONS; EXPERIMENTAL AIRFOIL (26a) - Concluded

(h) α = 3.5°

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.647	.807	.956	1.022	1.040							0.000
.003	-1.203	-.713	-.234	-.066	.031							.003
.012	-2.578	-2.025	-1.353	-1.139	-1.003							.012
.016	-2.483	-2.255	-1.565	-1.320	-1.191							.016
.029	-1.616	-2.343	-1.724	-1.488	-1.365							.029
.046	-1.347	-2.209	-1.732	-1.501	-1.385							.046
.071	-1.106	-.981	-1.691	-1.474	-1.362							.071
.098	-.927	-.910	-1.648	-1.449	-1.334							.098
.150	-.782	-.808	-1.554	-1.370	-1.285							.150
.200	-.699	-.718	-1.462	-1.326	-1.232							.200
.250	-.633	-.657	-.690	-1.312	-1.226							.250
.301	-.593	-.620	-.481	-1.302	-1.216							.301
.351	-.557	-.586	-.508	-1.278	-1.199							.351
.398	-.532	-.557	-.530	-1.248	-1.190							.398
.448	-.510	-.533	-.531	-.844	-1.187							.448
.499	-.486	-.510	-.529	-.442	-1.176							.499
.549	-.466	-.494	-.507	-.382	-1.160							.549
.600	-.460	-.479	-.504	-.411	-.787							.600
.652	-.443	-.463	-.486	-.413	-.443							.652
.700	-.432	-.443	-.471	-.424	-.375							.700
.750	-.407	-.425	-.449	-.414	-.335							.750
.801	-.390	-.394	-.421	-.404	-.315							.801
.850	-.365	-.355	-.394	-.371	-.297							.850
.899	-.286	-.272	-.282	-.275	-.216							.899
.932	-.228	-.207	-.214	-.200	-.149							.932
.972	-.082	-.068	-.050	-.040	-.015							.972
.989	-.026	-.011	.014	.025	.039							.989
1.000	.005	.010	.036	.057	.066							1.000
LOWER SURFACE												
.006	.993	.972	.942	.925	.918							.006
.011	.752	.722	.678	.646	.618							.011
.020	.562	.548	.503	.506	.480							.020
.029	.455	.421	.416	.392	.371							.029
.048	.309	.301	.276	.274	.269							.048
.069	.242	.233	.215	.203	.206							.069
.102	.166	.149	.142	.133	.139							.102
.151	.091	.077	.081	.079	.072							.151
.199	.043	.040	.032	.030	.034							.199
.251	.011	-.000	.001	-.004	-.009							.251
.299	.007	-.003	-.003	-.009	-.011							.299
.350	-.025	-.039	-.041	-.041	-.048							.350
.400	-.039	-.056	-.058	-.057	-.067							.400
.451	-.066	-.078	-.088	-.086	-.084							.451
.500	-.083	-.093	-.099	-.106	-.105							.500
.549	-.073	-.084	-.091	-.090	-.095							.549
.598	-.046	-.057	-.046	-.044	-.048							.598
.649	.023	.025	.044	.052	.056							.649
.699	.144	.150	.179	.189	.200							.699
.747	.246	.258	.291	.303	.313							.747
.800	.341	.350	.388	.403	.408							.800
.850	.402	.419	.451	.466	.473							.850
.899	.428	.449	.477	.498	.508							.899
.929	.422	.443	.478	.494	.504							.929
.950	.408	.429	.460	.480	.490							.950
.998	.003	.001	.030	.060	.066							.998

635 7

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL

(a) α = -0.5°

					CP AT -							
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
					UPPER SURFACE							
0.000	1.064	1.083	1.119	1.138	1.145	1.154	1.157	1.162	1.166	1.166	1.172	0.000
.003	.451	.508	.592	.630	.640	.668	.678	.691	.703	.710	.720	.003
.012	-.228	-.210	-.095	-.071	-.015	.034	.019	.036	.061	.080	.102	.012
.016	-.366	-.316	-.207	-.173	-.126	-.130	-.075	-.079	-.066	-.057	-.017	.016
.029	-.428	-.378	-.384	-.329	-.311	-.296	-.293	-.286	-.258	-.246	-.225	.029
.046	-.428	-.414	-.437	-.414	-.425	-.432	-.413	-.413	-.390	-.382	-.362	.046
.071	-.399	-.395	-.410	-.388	-.402	-.391	-.386	-.376	-.372	-.368	-.350	.071
.098	-.358	-.364	-.373	-.375	-.363	-.374	-.374	-.377	-.365	-.355	-.348	.098
.150	-.313	-.333	-.364	-.379	-.385	-.381	-.382	-.383	-.380	-.378	-.364	.150
.200	-.309	-.319	-.355	-.380	-.385	-.394	-.393	-.398	-.401	-.403	-.390	.200
.250	-.329	-.338	-.359	-.372	-.387	-.395	-.400	-.399	-.404	-.398	-.386	.250
.301	-.324	-.339	-.353	-.361	-.363	-.374	-.382	-.380	-.432	-.433	-.413	.301
.351	-.319	-.335	-.353	-.369	-.379	-.389	-.396	-.404	-.401	-.422	-.461	.351
.398	-.320	-.337	-.365	-.382	-.396	-.412	-.420	-.430	-.438	-.417	-.453	.398
.448	-.343	-.365	-.398	-.419	-.435	-.457	-.471	-.489	-.502	-.503	-.496	.448
.499	-.334	-.351	-.381	-.402	-.423	-.445	-.457	-.469	-.501	-.506	-.492	.499
.549	-.335	-.355	-.389	-.414	-.434	-.457	-.475	-.506	-.527	-.553	-.540	.549
.600	-.343	-.365	-.403	-.427	-.450	-.478	-.500	-.522	-.545	-.565	-.565	.600
.652	-.350	-.369	-.407	-.432	-.457	-.489	-.516	-.568	-.591	-.621	-.620	.652
.700	-.354	-.374	-.411	-.436	-.463	-.487	-.504	-.553	-.640	-.675	-.665	.700
.750	-.340	-.358	-.389	-.412	-.431	-.451	-.459	-.471	-.490	-.715	-.729	.750
.801	-.314	-.326	-.348	-.366	-.372	-.380	-.381	-.381	-.365	-.353	-.689	.801
.850	-.293	-.304	-.318	-.326	-.333	-.335	-.329	-.327	-.316	-.271	-.256	.850
.899	-.248	-.251	-.253	-.249	-.247	-.245	-.239	-.230	-.218	-.187	-.157	.899
.932	-.192	-.189	-.179	-.170	-.163	-.154	-.148	-.136	-.127	-.103	-.077	.932
.949	-.136	-.127	-.112	-.100	-.092	-.081	-.074	-.066	-.055	-.040	-.019	.949
.972	-.072	-.057	-.030	-.015	-.004	.006	.012	.018	.027	.034	.045	.972
.989	-.017	-.000	.026	.046	.054	.062	.068	.073	.079	.083	.082	.989
1.000	.044	.056	.076	.089	.094	.096	.101	.102	.108	.108	.100	1.000
					LOWER SURFACE							
.006	.462	.447	.498	.503	.512	.532	.525	.542	.545	.539	.556	.006
.011	.019	.018	.050	.070	.087	.098	.105	.104	.110	.128	.144	.011
.020	-.231	-.217	-.241	-.218	-.241	-.250	-.211	-.207	-.208	-.196	-.182	.020
.029	-.301	-.318	-.357	-.371	-.380	-.360	-.392	-.374	-.361	-.342	-.344	.029
.048	-.323	-.379	-.396	-.451	-.469	-.478	-.475	-.471	-.486	-.480	-.483	.048
.069	-.326	-.368	-.394	-.442	-.450	-.449	-.469	-.469	-.474	-.467	-.463	.069
.102	-.320	-.344	-.390	-.437	-.438	-.460	-.495	-.495	-.496	-.488	-.485	.102
.151	-.306	-.327	-.375	-.410	-.436	-.481	-.523	-.559	-.601	-.598	-.583	.151
.199	-.277	-.296	-.342	-.371	-.395	-.431	-.441	-.458	-.539	-.577	-.564	.199
.251	-.252	-.281	-.319	-.355	-.372	-.403	-.425	-.431	-.457	-.576	-.598	.251
.299	-.253	-.279	-.319	-.355	-.373	-.400	-.411	-.465	-.491	-.537	-.622	.299
.350	-.252	-.280	-.320	-.345	-.366	-.394	-.410	-.431	-.451	-.504	-.597	.350
.400	-.251	-.277	-.301	-.336	-.352	-.381	-.393	-.410	-.437	-.415	-.624	.400
.451	-.251	-.276	-.293	-.322	-.336	-.357	-.370	-.387	-.402	-.409	-.469	.451
.500	-.225	-.247	-.274	-.301	-.315	-.329	-.338	-.351	-.359	-.372	-.315	.500
.549	-.211	-.231	-.249	-.273	-.285	-.298	-.300	-.308	-.316	-.320	-.297	.549
.598	-.186	-.201	-.216	-.235	-.238	-.248	-.250	-.254	-.253	-.255	-.243	.598
.649	-.126	-.136	-.135	-.141	-.139	-.142	-.142	-.140	-.136	-.133	-.123	.649
.699	-.035	-.033	-.018	-.016	-.010	-.009	-.004	-.003	.006	.011	.019	.699
.747	.102	.110	.137	.144	.147	.156	.160	.161	.164	.171	.175	.747
.800	.243	.254	.277	.284	.286	.287	.290	.290	.292	.296	.297	.800
.850	.321	.329	.356	.359	.362	.362	.365	.361	.362	.367	.371	.850
.899	.334	.364	.390	.395	.399	.400	.404	.402	.404	.406	.412	.899
.929	.346	.362	.388	.397	.403	.406	.406	.407	.408	.414	.415	.929
.950	.329	.345	.374	.381	.386	.389	.396	.395	.399	.402	.403	.950
.969	.295	.309	.338	.346	.350	.356	.358	.360	.365	.368	.368	.969
.998	.037	.048	.073	.078	.083	.086	.092	.091	.099	.100	.092	.998

041 5

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(b) $\alpha = 0^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000					1.153	1.159	1.167	1.169	1.177	1.178	1.184	0.000
.003					.536	.576	.619	.627	.635	.653	.679	.003
.012					-.139	-.110	-.087	-.060	-.025	-.001	.016	.012
.016					-.269	-.218	-.183	-.165	-.145	-.137	-.107	.016
.029					-.404	-.404	-.369	-.378	-.342	-.308	-.289	.029
.046					-.538	-.531	-.506	-.496	-.485	-.476	-.467	.046
.071					-.493	-.474	-.479	-.466	-.460	-.439	-.427	.071
.098					-.457	-.454	-.454	-.444	-.428	-.419	-.404	.098
.150					-.424	-.434	-.429	-.430	-.417	-.418	-.417	.150
.200					-.414	-.433	-.427	-.438	-.426	-.433	-.425	.200
.250					-.413	-.431	-.433	-.428	-.423	-.413	-.426	.250
.301					-.431	-.448	-.444	-.520	-.503	-.480	-.455	.301
.351					-.412	-.433	-.435	-.427	-.524	-.514	-.489	.351
.398					-.427	-.450	-.460	-.463	-.458	-.512	-.508	.398
.448					-.453	-.481	-.495	-.511	-.490	-.546	-.543	.448
.499					-.452	-.473	-.488	-.525	-.524	-.525	-.556	.499
.549					-.458	-.487	-.505	-.532	-.563	-.574	-.581	.549
.600					-.476	-.512	-.527	-.559	-.585	-.598	-.614	.600
.652					-.474	-.511	-.543	-.592	-.638	-.650	-.666	.652
.700					-.469	-.500	-.517	-.568	-.672	-.672	-.719	.700
.750					-.436	-.455	-.461	-.468	-.575	-.736	-.775	.750
.801					-.379	-.383	-.384	-.379	-.345	-.360	-.670	.801
.850					-.333	-.334	-.330	-.325	-.298	-.264	-.243	.850
.899					-.250	-.244	-.233	-.226	-.208	-.178	-.146	.899
.932					-.161	-.155	-.146	-.136	-.120	-.096	-.071	.932
.949					-.092	-.083	-.071	-.066	-.051	-.035	-.018	.949
.972					-.009	-.002	.008	.013	.023	.033	.039	.972
.989					.048	.058	.062	.069	.074	.076	.074	.989
1.000					.083	.087	.089	.092	.096	.095	.089	1.000
LOWER SURFACE												
.006					.605	.592	.616	.610	.608	.614	.613	.006
.011					.197	.198	.169	.176	.195	.199	.197	.011
.020					-.134	-.121	-.119	-.121	-.132	-.122	-.110	.020
.029					-.266	-.280	-.285	-.264	-.275	-.257	-.260	.029
.048					-.377	-.361	-.387	-.362	-.376	-.387	-.388	.048
.069					-.361	-.375	-.393	-.404	-.405	-.401	-.394	.069
.102					-.370	-.407	-.408	-.430	-.430	-.436	-.429	.102
.151					-.375	-.408	-.426	-.453	-.514	-.530	-.539	.151
.199					-.347	-.379	-.386	-.415	-.433	-.447	-.520	.199
.251					-.338	-.366	-.379	-.397	-.424	-.439	-.529	.251
.299					-.320	-.339	-.355	-.361	-.417	-.457	-.479	.299
.350					-.330	-.351	-.372	-.384	-.413	-.426	-.472	.350
.400					-.325	-.345	-.363	-.375	-.402	-.428	-.483	.400
.451					-.313	-.331	-.344	-.357	-.371	-.393	-.431	.451
.500					-.291	-.308	-.321	-.327	-.339	-.354	-.358	.500
.549					-.267	-.279	-.287	-.293	-.301	-.309	-.318	.549
.598					-.226	-.238	-.238	-.243	-.247	-.250	-.252	.598
.649					-.132	-.134	-.132	-.134	-.134	-.135	-.131	.649
.699					-.006	-.005	-.003	.003	.005	.008	.012	.699
.747					.153	.156	.163	.168	.169	.174	.177	.747
.800					.296	.303	.305	.307	.307	.310	.309	.800
.850					.374	.377	.380	.381	.384	.384	.385	.850
.899					.412	.414	.418	.424	.424	.425	.425	.899
.929					.412	.417	.421	.427	.427	.430	.426	.929
.950					.394	.399	.405	.408	.410	.413	.412	.950
.969					.358	.360	.368	.373	.373	.378	.374	.969
.998					.073	.077	.082	.085	.085	.085	.075	.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(c) $\alpha = 0.5^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000					1.159	1.162	1.166	1.173	1.178	1.186	1.189	0.000
.003					.446	.470	.502	.513	.541	.563	.576	.003
.012					-.292	-.246	-.207	-.186	-.151	-.113	-.093	.012
.016					-.434	-.391	-.349	-.312	-.280	-.245	-.220	.016
.029					-.579	-.526	-.478	-.463	-.424	-.409	-.379	.029
.046					-.757	-.698	-.688	-.660	-.631	-.586	-.560	.046
.071					-.614	-.665	-.672	-.644	-.623	-.592	-.541	.071
.098					-.559	-.575	-.606	-.630	-.646	-.619	-.603	.098
.150					-.472	-.467	-.460	-.478	-.448	-.480	-.456	.150
.200					-.474	-.483	-.497	-.493	-.489	-.481	-.482	.200
.250					-.472	-.476	-.470	-.450	-.467	-.519	-.498	.250
.301					-.503	-.573	-.579	-.549	-.512	-.533	-.518	.301
.351					-.465	-.478	-.583	-.600	-.570	-.528	-.538	.351
.398					-.476	-.494	-.480	-.601	-.580	-.547	-.535	.398
.448					-.487	-.523	-.532	-.619	-.593	-.603	-.588	.448
.499					-.463	-.512	-.529	-.599	-.615	-.605	-.607	.499
.549					-.465	-.522	-.553	-.615	-.640	-.622	-.640	.549
.600					-.494	-.529	-.553	-.549	-.661	-.648	-.637	.600
.652					-.489	-.534	-.572	-.574	-.704	-.712	-.708	.652
.700					-.481	-.508	-.532	-.589	-.735	-.759	-.755	.700
.750					-.441	-.453	-.466	-.459	-.475	-.816	-.818	.750
.801					-.375	-.384	-.386	-.375	-.328	-.416	-.704	.801
.850					-.326	-.327	-.326	-.315	-.288	-.234	-.250	.850
.899					-.243	-.230	-.226	-.217	-.193	-.149	-.131	.899
.932					-.157	-.142	-.138	-.128	-.112	-.075	-.061	.932
.949					-.085	-.071	-.062	-.059	-.046	-.022	-.019	.949
.972					-.006	.005	.010	.015	.028	.036	.029	.972
.989					.049	.057	.058	.065	.068	.074	.052	.989
1.000					.076	.079	.082	.086	.088	.088	.067	1.000
LOWER SURFACE												
.006					.712	.705	.698	.707	.705	.696	.691	.006
.011					.297	.304	.293	.307	.302	.298	.305	.011
.020					.004	.008	-.012	.005	-.002	-.009	-.006	.020
.029					-.126	-.123	-.145	-.143	-.145	-.150	-.157	.029
.048					-.221	-.242	-.270	-.251	-.271	-.261	-.271	.048
.069					-.237	-.268	-.266	-.266	-.275	-.293	-.302	.069
.102					-.286	-.295	-.305	-.316	-.323	-.339	-.363	.102
.151					-.294	-.316	-.329	-.345	-.362	-.391	-.420	.151
.199					-.281	-.293	-.309	-.331	-.343	-.355	-.380	.199
.251					-.280	-.295	-.309	-.331	-.338	-.366	-.380	.251
.299					-.265	-.282	-.294	-.310	-.314	-.342	-.428	.299
.350					-.278	-.301	-.313	-.331	-.350	-.380	-.392	.350
.400					-.281	-.301	-.313	-.331	-.349	-.372	-.411	.400
.451					-.276	-.291	-.298	-.314	-.330	-.348	-.383	.451
.500					-.262	-.275	-.288	-.303	-.306	-.320	-.341	.500
.549					-.240	-.258	-.264	-.271	-.279	-.284	-.306	.549
.598					-.205	-.217	-.223	-.226	-.227	-.235	-.243	.598
.649					-.121	-.120	-.122	-.123	-.121	-.124	-.125	.649
.699					.001	.003	.008	.009	.015	.013	.013	.699
.747					.162	.165	.169	.176	.176	.179	.178	.747
.800					.307	.311	.313	.316	.319	.318	.307	.800
.850					.385	.386	.388	.393	.393	.394	.381	.850
.899					.419	.425	.425	.431	.433	.432	.423	.899
.929					.417	.425	.425	.426	.432	.433	.426	.929
.950					.400	.406	.406	.410	.415	.416	.408	.950
.969					.361	.366	.372	.372	.373	.375	.368	.969
.998					.068	.068	.072	.074	.078	.072	.058	.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(d) $\alpha = 1.0^{o}$

	CP AT -											
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.76	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	1.025	1.077	1.115	1.137	1.150	1.159	1.168	1.177	1.178	1.183	1.189	0.000
.003	-.092	.048	.184	.293	.328	.379	.410	.443	.453	.477	.524	.003
.012	-.776	-.722	-.631	-.475	-.405	-.374	-.315	-.280	-.247	-.185	-.142	.012
.016	-.872	-.824	-.767	-.663	-.570	-.504	-.463	-.419	-.392	-.354	-.294	.016
.029	-.794	-.803	-.831	-.751	-.695	-.638	-.607	-.567	-.522	-.481	-.450	.029
.046	-.746	-.776	-.901	-.907	-.839	-.778	-.753	-.728	-.698	-.653	-.621	.046
.071	-.651	-.660	-.730	-.821	-.867	-.811	-.773	-.753	-.712	-.673	-.639	.071
.098	-.539	-.554	-.615	-.680	-.807	-.857	-.821	-.808	-.771	-.740	-.701	.098
.150	-.458	-.479	-.517	-.537	-.496	-.727	-.747	-.743	-.715	-.662	-.632	.150
.200	-.438	-.454	-.503	-.525	-.535	-.467	-.706	-.704	-.661	-.645	-.649	.200
.250	-.420	-.446	-.484	-.505	-.519	-.467	-.457	-.625	-.675	-.613	-.610	.250
.301	-.414	-.442	-.495	-.524	-.563	-.612	-.554	-.633	-.652	-.614	-.601	.301
.351	-.391	[illegible]	-.460	-.487	-.503	-.643	-.560	-.612	-.658	-.638	-.622	.351
.398	-.393	-.414	-.458	-.487	-.507	-.511	-.581	-.563	-.625	-.624	-.602	.398
.448	-.409	-.426	-.477	-.511	-.532	-.554	-.590	-.589	-.674	-.677	-.659	.448
.499	-.449	-.470	-.515	-.552	-.576	-.605	-.683	-.567	-.736	-.747	-.741	.499
.549	-.381	-.407	-.449	-.483	-.508	-.551	-.614	-.588	-.687	-.704	-.698	.549
.600	-.392	-.412	-.460	-.491	-.518	-.560	-.573	-.607	-.632	-.719	-.727	.600
.652	-.396	-.420	-.463	-.493	-.517	-.571	-.601	-.674	-.691	-.750	-.776	.652
.700	-.395	-.411	-.459	-.484	-.497	-.538	-.540	-.722	-.752	-.769	-.759	.700
.750	-.369	-.386	-.425	-.441	-.452	-.467	-.469	-.459	-.685	-.809	-.805	.750
.801	-.335	-.355	-.372	-.382	-.391	-.391	-.391	-.365	-.330	-.406	-.644	.801
.850	-.309	-.317	-.334	-.331	-.331	-.334	-.326	-.314	-.279	-.241	-.254	.850
.899	-.254	-.256	-.251	-.243	-.239	-.233	-.225	-.215	-.192	-.140	-.131	.899
.932	-.191	-.186	-.172	-.152	-.152	-.141	-.132	-.125	-.105	-.071	-.066	.932
.949	-.134	-.125	-.104	-.090	-.080	-.072	-.065	-.057	-.043	-.022	-.027	.949
.972	-.068	-.051	-.027	-.014	-.006	.004	.012	.017	.029	.030	.017	.972
.989	-.012	.000	.025	.040	.043	.055	.063	.066	.071	.071	.041	.989
1.000	.039	.050	.063	.066	.075	.078	.086	.089	.089	.081	.059	1.000
LOWER SURFACE												
.006	.791	.792	.802	.786	.786	.776	.788	.782	.768	.764	.756	.006
.011	.423	.411	.423	.402	.410	.406	.407	.394	.401	.406	.381	.011
.020	.134	.129	.159	.125	.109	.136	.122	.110	.105	.080	.069	.020
.029	.037	.010	.014	.007	-.026	-.032	-.027	-.040	-.036	-.040	-.068	.029
.048	-.072	-.088	-.094	-.118	-.133	-.140	-.141	-.149	-.166	-.174	-.178	.048
.069	-.094	-.106	-.123	-.143	-.175	-.183	-.176	-.175	-.184	-.191	-.207	.069
.102	-.131	-.143	-.176	-.194	-.193	-.215	-.216	-.219	-.234	-.247	-.255	.102
.151	-.150	-.174	-.199	-.216	-.227	-.239	-.251	-.256	-.276	-.284	-.305	.151
.199	-.155	-.175	-.200	-.218	-.230	-.241	-.249	-.249	-.266	-.282	-.309	.199
.251	-.155	-.176	-.200	-.218	-.231	-.243	-.250	-.256	-.282	-.294	-.311	.251
.299	-.152	-.150	-.187	-.205	-.217	-.230	-.242	-.246	-.261	-.269	-.286	.299
.350	-.158	-.174	-.205	-.225	-.234	-.256	-.265	-.274	-.286	-.307	-.346	.350
.400	-.157	-.174	-.206	-.227	-.240	-.255	-.265	-.280	-.288	-.311	-.354	.400
.451	-.167	-.183	-.210	-.231	-.239	-.252	-.264	-.269	-.284	-.295	-.341	.451
.500	-.165	-.177	-.205	-.221	-.231	-.244	-.249	-.258	-.274	-.282	-.314	.500
.549	-.154	-.169	-.191	-.207	-.218	-.224	-.230	-.234	-.246	-.260	-.275	.549
.598	-.139	-.150	-.168	-.179	-.183	-.193	-.197	-.198	-.209	-.209	-.225	.598
.649	-.090	-.089	-.099	-.102	-.106	-.104	-.100	-.107	-.104	-.106	-.120	.649
.699	-.003	-.001	.008	.011	.013	.018	.017	.024	.021	.028	.017	.699
.747	.126	.141	.154	.165	.169	.171	.181	.185	.183	.190	.177	.747
.800	.268	.284	.302	.310	.312	.319	.319	.325	.326	.328	.318	.800
.850	.345	.363	.383	.390	.390	.398	.395	.401	.400	.405	.398	.850
.899	.374	.394	.412	.423	.429	.431	.435	.439	.442	.443	.432	.899
.929	.365	.385	.410	.420	.425	.429	.433	.436	.441	.443	.434	.929
.950	.343	.363	.389	.401	.405	.409	.418	.415	.419	.428	.411	.950
.969	.302	.323	.348	.357	.361	.369	.370	.376	.378	.380	.365	.969
.998	.027	.040	.053	.061	.057	.068	.072	.078	.078	.067	.045	.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(e) $\alpha = 1.5^{\circ}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.978	1.043	1.091	1.122	1.136	1.155	1.160	1.167	1.174	1.176		0.000
.003	-.348	-.207	.031	.145	.218	.291	.316	.348	.371	.421		.003
.012	-1.046	-.993	-.761	-.635	-.548	-.463	-.417	-.390	-.343	-.272		.012
.016	-1.103	-1.108	-.949	-.770	-.696	-.604	-.569	-.551	-.473	-.440		.016
.029	-.965	-.985	-1.014	-.927	-.835	-.755	-.742	-.680	-.638	-.572		.029
.046	-.860	-.912	-1.098	-1.010	-.972	-.878	-.849	-.786	-.756	-.707		.046
.071	-.709	-.771	-.938	-1.005	-.951	-.876	-.834	-.787	-.764	-.716		.071
.098	-.611	-.632	-.630	-.996	-.988	-.937	-.892	-.864	-.833	-.794		.098
.150	-.511	-.544	-.580	-.446	-.895	-.875	-.847	-.816	-.791	-.745		.150
.200	-.488	-.511	-.556	-.560	-.684	-.855	-.853	-.810	-.778	-.743		.200
.250	-.460	-.487	-.524	-.552	-.487	-.813	-.804	-.771	-.750	-.715		.250
.301	-.450	-.479	-.530	-.579	-.601	-.757	-.763	-.765	-.740	-.711		.301
.351	-.423	-.449	-.494	-.526	-.526	-.701	-.733	-.775	-.769	-.730		.351
.398	-.421	-.443	-.488	-.528	-.535	-.427	-.719	-.713	-.718	-.703		.398
.448	-.420	-.445	-.492	-.529	-.553	-.521	-.751	-.762	-.758	-.723		.448
.499	-.410	-.433	-.479	-.519	-.541	-.550	-.661	-.779	-.769	-.748		.499
.549	-.404	-.431	-.473	-.502	-.534	-.586	-.438	-.802	-.796	-.780		.549
.600	-.404	-.434	-.468	-.505	-.534	-.580	-.512	-.810	-.820	-.805		.600
.652	-.396	-.420	-.464	-.496	-.521	-.566	-.558	-.544	-.861	-.854		.652
.700	-.397	-.419	-.457	-.483	-.503	-.538	-.578	-.434	-.896	-.894		.700
.750	-.374	-.395	-.422	-.445	-.461	-.479	-.479	-.390	-.468	-.936		.750
.801	-.338	-.355	-.372	-.384	-.386	-.400	-.391	-.339	-.296	-.373		.801
.850	-.311	-.323	-.329	-.332	-.327	-.339	-.335	-.300	-.235	-.237		.850
.899	-.254	-.259	-.244	-.242	-.233	-.237	-.230	-.211	-.161	-.135		.899
.932	-.192	-.188	-.168	-.155	-.151	-.148	-.140	-.120	-.085	-.072		.932
.949	-.131	-.127	-.103	-.089	-.078	-.075	-.068	-.053	-.027	-.030		.949
.972	-.065	-.052	-.026	-.010	-.001	.005	.012	.025	.036	.019		.972
.989	-.015	.000	.033	.039	.052	.057	.068	.076	.071	.044		.989
1.000	.035	.047	.063	.069	.076	.091	.094	.102	.094	.053		1.000
LOWER SURFACE												
.006	.867	.861	.873	.855	.858	.850	.851	.854	.829	.833		.006
.011	.539	.544	.530	.514	.499	.481	.485	.475	.475	.450		.011
.020	.264	.252	.238	.231	.229	.213	.208	.213	.200	.166		.020
.029	.130	.124	.110	.076	.079	.062	.067	.082	.049	.048		.029
.048	.023	-.002	-.014	-.025	-.046	-.057	-.039	-.053	-.067	-.089		.048
.069	-.027	-.038	-.049	-.072	-.072	-.093	-.084	-.102	-.105	-.121		.069
.102	-.070	-.082	-.101	-.113	-.120	-.138	-.144	-.151	-.162	-.175		.102
.151	-.109	-.119	-.140	-.157	-.166	-.172	-.185	-.185	-.200	-.219		.151
.199	-.107	-.117	-.143	-.157	-.172	-.178	-.184	-.184	-.199	-.218		.199
.251	-.107	-.134	-.153	-.178	-.179	-.192	-.196	-.204	-.215	-.241		.251
.299	-.111	-.132	-.147	-.167	-.176	-.188	-.195	-.203	-.215	-.237		.299
.350	-.134	-.147	-.174	-.192	-.201	-.217	-.220	-.229	-.243	-.275		.350
.400	-.142	-.155	-.177	-.196	-.206	-.218	-.223	-.230	-.248	-.279		.400
.451	-.148	-.157	-.181	-.202	-.207	-.223	-.225	-.230	-.247	-.272		.451
.500	-.144	-.158	-.179	-.196	-.206	-.215	-.218	-.220	-.235	-.267		.500
.549	-.138	-.147	-.165	-.184	-.192	-.193	-.204	-.209	-.218	-.243		.549
.598	-.123	-.136	-.146	-.151	-.164	-.168	-.167	-.169	-.178	-.199		.598
.649	-.076	-.085	-.080	-.083	-.086	-.084	-.083	-.077	-.083	-.098		.649
.699	.008	.013	.018	.024	.025	.032	.037	.040	.039	.027		.699
.747	.132	.144	.161	.173	.178	.185	.192	.196	.194	.191		.747
.800	.273	.287	.311	.315	.318	.328	.331	.333	.336	.331		.800
.850	.349	.367	.389	.391	.399	.402	.404	.408	.410	.406		.850
.899	.380	.394	.419	.425	.434	.438	.438	.450	.443	.441		.899
.929	.367	.388	.417	.417	.428	.436	.441	.443	.445	.440		.929
.950	.346	.364	.391	.401	.404	.414	.419	.426	.426	.416		.950
.969	.307	.324	.349	.353	.362	.375	.376	.385	.388	.372		.969
.998	.027	.035	.046	.059	.065	.075	.082	.093	.084	.050		.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(f) $\alpha = 2.0^{o}$

CP AT -												
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.907	.974	1.074	1.099	1.121	1.138	1.145	1.154	1.167	1.175		0.000
.003	-.549	-.338	-.063	.064	.142	.211	.239	.275	.352	.379		.003
.012	-1.303	-1.219	-.874	-.710	-.650	-.554	-.505	-.444	-.395	-.354		.012
.016	-1.327	-1.339	-1.080	-.906	-.824	-.701	-.670	-.607	-.560	-.504		.016
.029	-1.123	-1.187	-1.176	-1.033	-.958	-.862	-.828	-.741	-.721	-.664		.029
.046	-.989	-1.080	-1.320	-1.164	-1.090	-.995	-.953	-.904	-.846	-.791		.046
.071	-.772	-.841	-1.184	-1.114	-1.056	-.954	-.912	-.858	-.818	-.770		.071
.098	-.682	-.708	-.752	-1.139	-1.087	-1.016	-.962	-.921	-.873	-.834		.098
.150	-.561	-.602	-.622	-1.028	-1.010	-.956	-.910	-.860	-.831	-.796		.150
.200	-.525	-.559	-.597	-.447	-1.000	-.937	-.904	-.872	-.829	-.798		.200
.250	-.512	-.544	-.585	-.530	-.946	-.922	-.889	-.860	-.822	-.777		.250
.301	-.484	-.514	-.573	-.576	-.850	-.881	-.861	-.842	-.811	-.769		.301
.351	-.456	-.490	-.531	-.543	-.393	-.896	-.878	-.852	-.822	-.799		.351
.398	-.444	-.477	-.526	-.547	-.466	-.827	-.836	-.823	-.786	-.774		.398
.448	-.452	-.488	-.542	-.574	-.541	-.902	-.894	-.874	-.849	-.822		.448
.499	-.427	-.460	-.506	-.540	-.541	-.859	-.864	-.858	-.844	-.815		.499
.549	-.422	-.451	-.502	-.527	-.551	-.454	-.880	-.680	-.863	-.836		.549
.600	-.424	-.450	-.498	-.527	-.565	-.433	-.886	-.890	-.876	-.849		.600
.652	-.418	-.441	-.486	-.510	-.541	-.495	-.752	-.937	-.918	-.901		.652
.700	-.413	-.435	-.473	-.497	-.526	-.495	-.381	-.970	-.963	-.940		.700
.750	-.388	-.404	-.438	-.452	-.473	-.464	-.356	-.428	-.880	-.797		.750
.801	-.350	-.360	-.377	-.383	-.401	-.387	-.336	-.283	-.371	-.351		.801
.850	-.316	-.326	-.323	-.333	-.341	-.333	-.298	-.222	-.241	-.261		.850
.899	-.260	-.261	-.241	-.241	-.250	-.238	-.216	-.151	-.136	-.178		.899
.932	-.194	-.180	-.161	-.153	-.160	-.144	-.125	-.084	-.072	-.133		.932
.949	-.134	-.123	-.097	-.083	-.086	-.073	-.061	-.028	-.030	-.102		.949
.972	-.066	-.050	-.022	-.004	.000	.012	.018	.032	.010	-.067		.972
.989	-.013	.003	.028	.042	.052	.064	.074	.071	.038	-.014		.989
1.000	.033	.047	.059	.070	.083	.096	.103	.093	.050	.033		1.000
LOWER SURFACE												
.006	.923	.929	.919	.922	.918	.907	.900	.888	.876	.872		.006
.011	.625	.608	.611	.594	.590	.574	.568	.560	.521	.510		.011
.020	.368	.353	.324	.308	.328	.303	.291	.285	.265	.250		.020
.029	.222	.208	.188	.178	.184	.152	.160	.144	.130	.107		.029
.048	.092	.080	.050	.050	.052	.045	.031	-.002	.008	-.019		.048
.069	.040	.042	.034	.007	.007	-.009	-.006	-.018	-.034	-.063		.069
.102	-.017	-.023	-.043	-.051	-.049	-.067	-.065	-.083	-.090	-.118		.102
.151	-.056	-.068	-.088	-.100	-.098	-.109	-.114	-.132	-.150	-.170		.151
.199	-.076	-.070	-.088	-.091	-.100	-.111	-.105	-.124	-.135	-.166		.199
.251	-.093	-.084	-.105	-.116	-.116	-.130	-.132	-.140	-.161	-.189		.251
.299	-.083	-.113	-.126	-.142	-.142	-.157	-.156	-.170	-.188	-.211		.299
.350	-.106	-.118	-.140	-.158	-.157	-.169	-.175	-.190	-.196	-.235		.350
.400	-.117	-.133	-.153	-.165	-.169	-.179	-.178	-.194	-.218	-.237		.400
.451	-.122	-.138	-.158	-.166	-.171	-.183	-.183	-.196	-.224	-.236		.451
.500	-.122	-.137	-.159	-.171	-.168	-.178	-.189	-.189	-.213	-.240		.500
.549	-.116	-.130	-.147	-.159	-.160	-.169	-.167	-.180	-.191	-.219		.549
.598	-.107	-.115	-.132	-.133	-.139	-.139	-.141	-.147	-.160	-.189		.598
.649	-.064	-.064	-.070	-.070	-.066	-.061	-.061	-.066	-.082	-.095		.649
.699	.016	.023	.028	.038	.045	.045	.052	.052	.038	.024		.699
.747	.139	.152	.173	.182	.192	.199	.203	.203	.196	.183		.747
.800	.277	.291	.312	.327	.332	.338	.342	.340	.335	.318		.800
.850	.353	.364	.388	.397	.400	.412	.412	.412	.403	.390		.850
.899	.383	.392	.416	.429	.437	.443	.448	.446	.440	.422		.899
.929	.373	.382	.407	.424	.433	.443	.445	.445	.439	.419		.929
.950	.348	.366	.384	.400	.409	.422	.428	.428	.417	.392		.950
.969	.309	.319	.343	.360	.366	.380	.388	.386	.374	.344		.969
.998	.023	.040	.047	.057	.069	.082	.094	.083	.046	-.010		.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS; THEORETICAL AIRFOIL - Continued

(g) $a = 2.5^{o}$

	CP AT -											
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
						UPPER SURFACE						
0.000	.813	.897	1.016	1.070	1.089	1.116	1.128	1.135				0.000
.003	-.906	-.682	-.288	-.114	-.041	.027	.080	.113				.003
.012	-1.587	-1.488	-1.046	-.832	-.741	-.640	-.596	-.544				.012
.016	-1.568	-1.636	-1.220	-1.007	-.913	-.802	-.762	-.701				.016
.029	-1.304	-1.435	-1.389	-1.196	-1.081	-.982	-.925	-.865				.029
.046	-1.091	-1.230	-1.447	-1.260	-1.157	-1.058	-1.033	-.971				.046
.071	-.879	-.932	-1.382	-1.221	-1.146	-1.037	-1.002	-.956				.071
.098	-.757	-.803	-1.307	-1.258	-1.200	-1.093	-1.040	-1.014				.098
.150	-.619	-.656	-.500	-1.159	-1.095	-1.015	-.971	-.948				.150
.200	-.580	-.606	-.600	-1.138	-1.100	-1.007	-.986	-.923				.200
.250	-.551	-.584	-.609	-1.100	-1.068	-1.008	-.979	-.942				.250
.301	-.518	-.552	-.594	-.418	-1.014	-.994	-.955	-.920				.301
.351	-.487	-.516	-.563	-.448	-.991	-.950	-.929	-.897				.351
.398	-.475	-.505	-.549	-.494	-.996	-.959	-.936	-.904				.398
.448	-.481	-.515	-.557	-.545	-.477	-.988	-.969	-.931				.448
.499	-.454	-.482	-.524	-.534	-.412	-.973	-.960	-.938				.499
.549	-.444	-.471	-.514	-.541	-.465	-.997	-.974	-.954				.549
.600	-.439	-.465	-.509	-.536	-.503	-.995	-.983	-.966				.600
.652	-.426	-.451	-.494	-.524	-.504	-.567	-1.024	-1.004				.652
.700	-.423	-.444	-.485	-.517	-.506	-.360	-.728	-1.040				.700
.750	-.391	-.411	-.443	-.466	-.465	-.340	-.391	-.545				.750
.801	-.351	-.368	-.385	-.400	-.393	-.308	-.282	-.357				.801
.850	-.318	-.329	-.330	-.345	-.340	-.279	-.226	-.242				.850
.899	-.261	-.257	-.241	-.251	-.246	-.202	-.161	-.136				.899
.932	-.191	-.181	-.152	-.166	-.159	-.132	-.086	-.076				.932
.949	-.130	-.121	-.098	-.091	-.083	-.066	-.036	-.035				.949
.972	-.065	-.046	-.019	-.011	-.002	.014	.029	.013				.972
.989	-.009	-.000	.032	.048	.050	.067	.071	.036				.989
1.000	.035	.044	.056	.075	.089	.104	.097	.058				.000
						LOWER SURFACE						
.006	.985	.983	.989	.967	.964	.957	.956	.941				.006
.011	.708	.705	.689	.668	.660	.653	.643	.627				.011
.020	.464	.434	.431	.409	.398	.387	.378	.364				.020
.029	.307	.299	.290	.271	.265	.247	.246	.220				.029
.048	.174	.167	.149	.141	.125	.125	.113	.100				.048
.069	.114	.113	.086	.084	.071	.058	.061	.043				.069
.102	.039	.033	.024	.010	.017	.003	.001	-.011				.102
.151	-.009	-.013	-.035	-.035	-.034	-.048	-.057	-.049				.151
.199	-.031	-.036	-.049	-.062	-.056	-.067	-.076	-.069				.199
.251	-.034	-.068	-.074	-.079	-.075	-.091	-.096	-.095				.251
.299	-.050	-.054	-.073	-.078	-.092	-.087	-.097	-.125				.299
.350	-.075	-.084	-.103	-.108	-.114	-.119	-.124	-.137				.350
.400	-.089	-.104	-.119	-.123	-.128	-.133	-.145	-.165				.400
.451	-.102	-.114	-.129	-.136	-.135	-.147	-.149	-.168				.451
.500	-.107	-.113	-.127	-.133	-.138	-.145	-.147	-.162				.500
.549	-.106	-.114	-.126	-.130	-.128	-.133	-.135	-.148				.549
.598	-.090	-.099	-.111	-.115	-.116	-.107	-.115	-.133				.598
.649	-.049	-.052	-.055	-.047	-.046	-.041	-.046	-.058				.649
.699	.029	.035	.040	.055	.060	.067	.059	.050				.699
.747	.148	.160	.176	.195	.203	.216	.213	.205				.747
.800	.287	.304	.325	.337	.345	.358	.355	.344				.800
.850	.359	.378	.400	.414	.423	.433	.434	.421				.850
.899	.386	.402	.430	.448	.453	.466	.468	.453				.899
.929	.377	.396	.426	.439	.445	.463	.462	.450				.929
.950	.352	.376	.402	.414	.422	.436	.442	.424				.950
.969	.315	.330	.352	.370	.385	.394	.399	.375				.969
.998	.021	.035	.039	.058	.076	.089	.086	.039				.998

TABLE III.- SURFACE PRESSURE DISTRIBUTIONS: THEORETICAL AIRFOIL - Concluded

(h) $\alpha = 3.5^{\circ}$

	CP AT -											
X/C	M=0.50	M=0.60	M=0.70	M=0.74	M=0.76	M=0.78	M=0.79	M=0.80	M=0.81	M=0.82	M=0.83	X/C
UPPER SURFACE												
0.000	.563	.755	.926	.985	1.024							0.000
.003	-1.424	-.983	-.496	-.280	-.164							.003
.012	-2.177	-1.880	-1.243	-1.026	-.913							.012
.016	-2.111	-2.056	-1.424	-1.160	-1.057							.016
.029	-1.771	-2.124	-1.641	-1.378	-1.283							.029
.046	-1.313	-2.014	-1.574	-1.408	-1.316							.046
.071	-1.089	-1.007	-1.615	-1.400	-1.294							.071
.098	-.896	-.923	-1.629	-1.423	-1.319							.098
.150	-.730	-.770	-1.507	-1.344	-1.243							.150
.200	-.668	-.695	-1.418	-1.303	-1.220							.200
.250	-.624	-.656	-.464	-1.280	-1.206							.250
.301	-.582	-.620	-.497	-1.229	-1.184							.301
.351	-.545	-.575	-.536	-1.253	-1.190							.351
.398	-.522	-.561	-.552	-1.168	-1.137							.398
.448	-.527	-.569	-.581	-1.071	-1.196							.448
.499	-.489	-.518	-.544	-.406	-1.147							.499
.549	-.475	-.501	-.537	-.396	-1.173							.549
.600	-.466	-.495	-.531	-.428	-.742							.600
.652	-.454	-.481	-.519	-.455	-.445							.652
.700	-.446	-.464	-.488	-.460	-.354							.700
.750	-.411	-.428	-.451	-.429	-.331							.750
.801	-.363	-.376	-.388	-.379	-.307							.801
.850	-.321	-.330	-.337	-.326	-.277							.850
.899	-.257	-.255	-.253	-.247	-.214							.899
.932	-.187	-.180	-.167	-.164	-.130							.932
.949	-.124	-.114	-.102	-.096	-.063							.949
.972	-.058	-.045	-.023	-.016	.005							.972
.989	-.011	.007	.028	.042	.059							.989
1.000	.035	.039	.058	.087	.100							1.000
LOWER SURFACE												
.006	1.044	1.055	1.055	1.054	1.046							.006
.011	.853	.833	.829	.793	.781							.011
.020	.615	.601	.571	.574	.552							.020
.029	.456	.453	.440	.414	.406							.029
.048	.312	.304	.279	.265	.275							.048
.069	.237	.225	.229	.205	.211							.069
.102	.147	.142	.136	.131	.125							.102
.151	.076	.074	.074	.077	.062							.151
.199	.044	.038	.044	.042	.032							.199
.251	.022	.010	.008	.006	.006							.251
.299	.020	.005	.003	.004	-.004							.299
.350	-.022	-.033	-.034	-.032	-.042							.350
.400	-.037	-.048	-.056	-.057	-.057							.400
.451	-.053	-.068	-.071	-.076	-.077							.451
.500	-.062	-.079	-.079	-.074	-.084							.500
.549	-.058	-.069	-.076	-.071	-.076							.549
.598	-.057	-.059	-.067	-.062	-.062							.598
.649	-.019	-.023	-.017	-.011	-.002							.649
.699	.049	.052	.066	.084	.085							.699
.747	.168	.179	.203	.220	.230							.747
.800	.302	.316	.344	.362	.371							.800
.850	.377	.392	.421	.436	.443							.850
.899	.397	.416	.447	.469	.472							.899
.929	.388	.404	.436	.458	.461							.929
.950	.364	.377	.409	.434	.443							.950
.969	.317	.333	.365	.383	.398							.969
.998	.019	.018	.048	.078	.091							.998

[redacted]

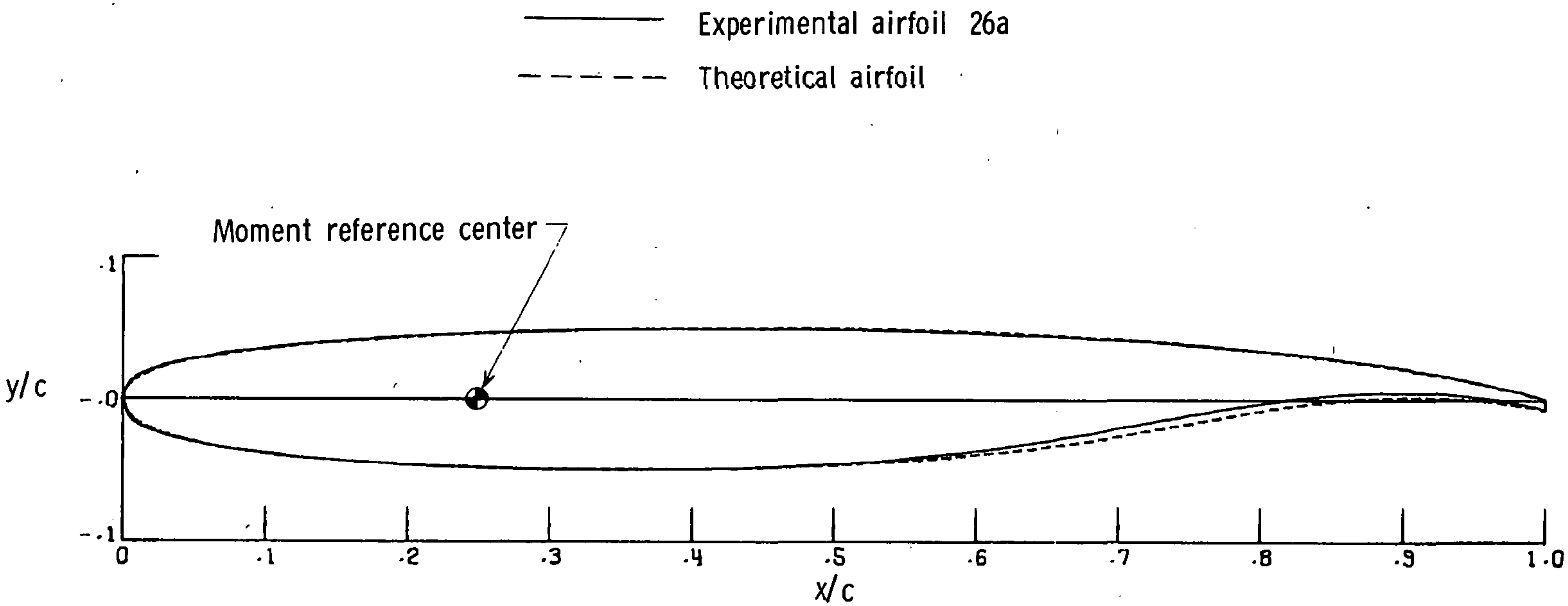
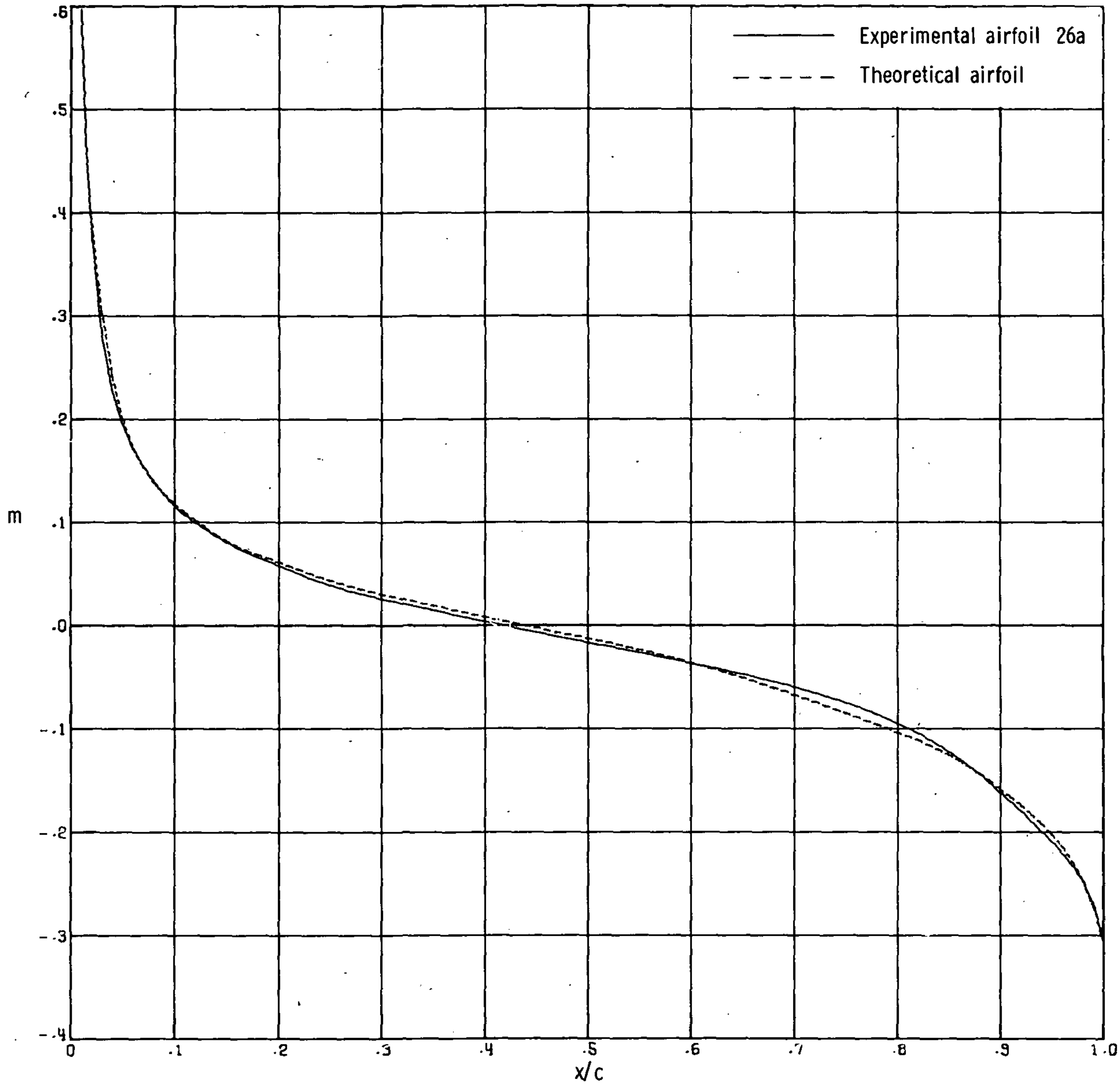


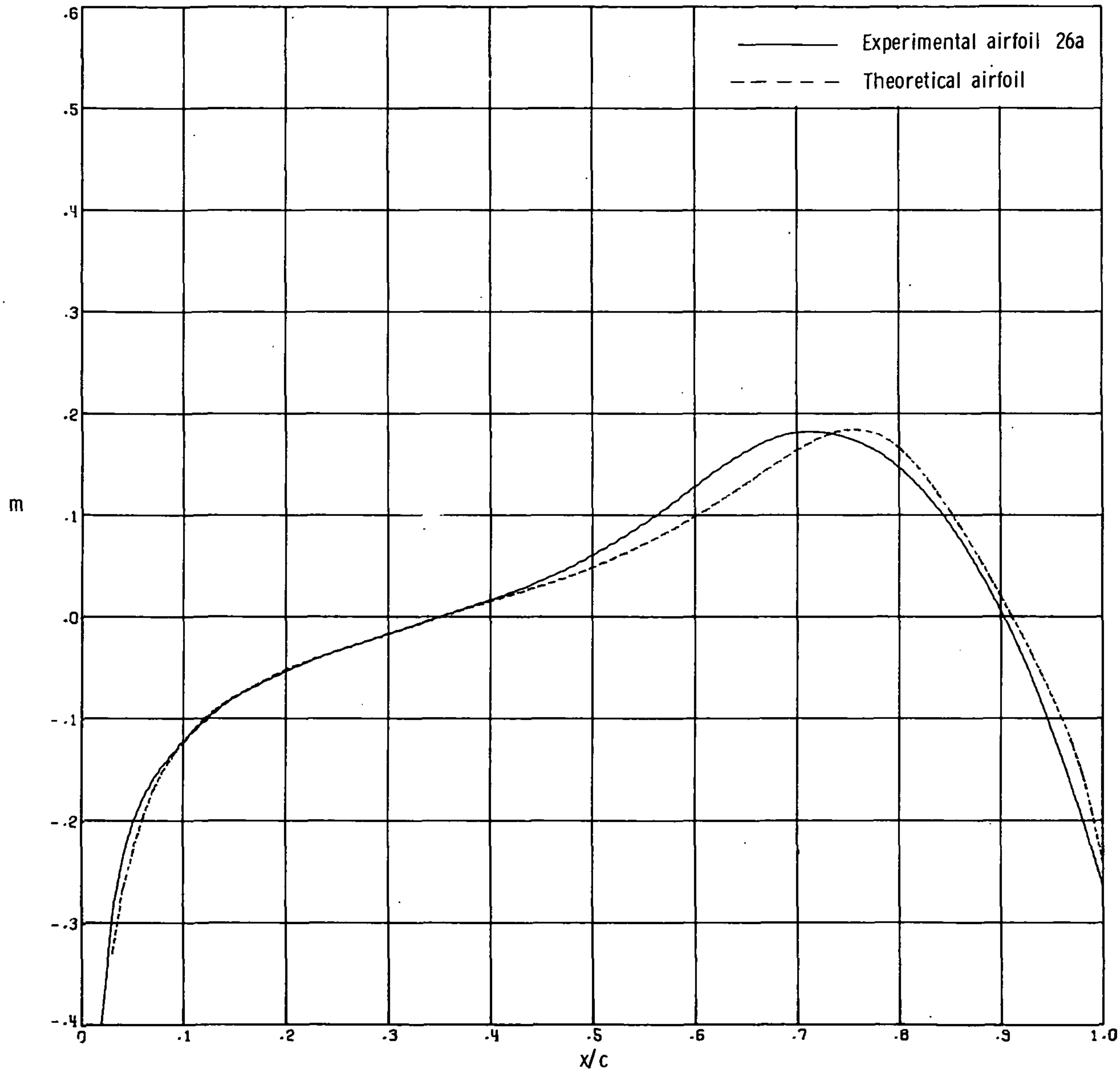
Figure 1.- Airfoil sketches.

[redacted]



(a) Upper surface.

Figure 2.- Chordwise distribution of airfoil surface slopes.



(b) Lower surface.

Figure 2.- Concluded.

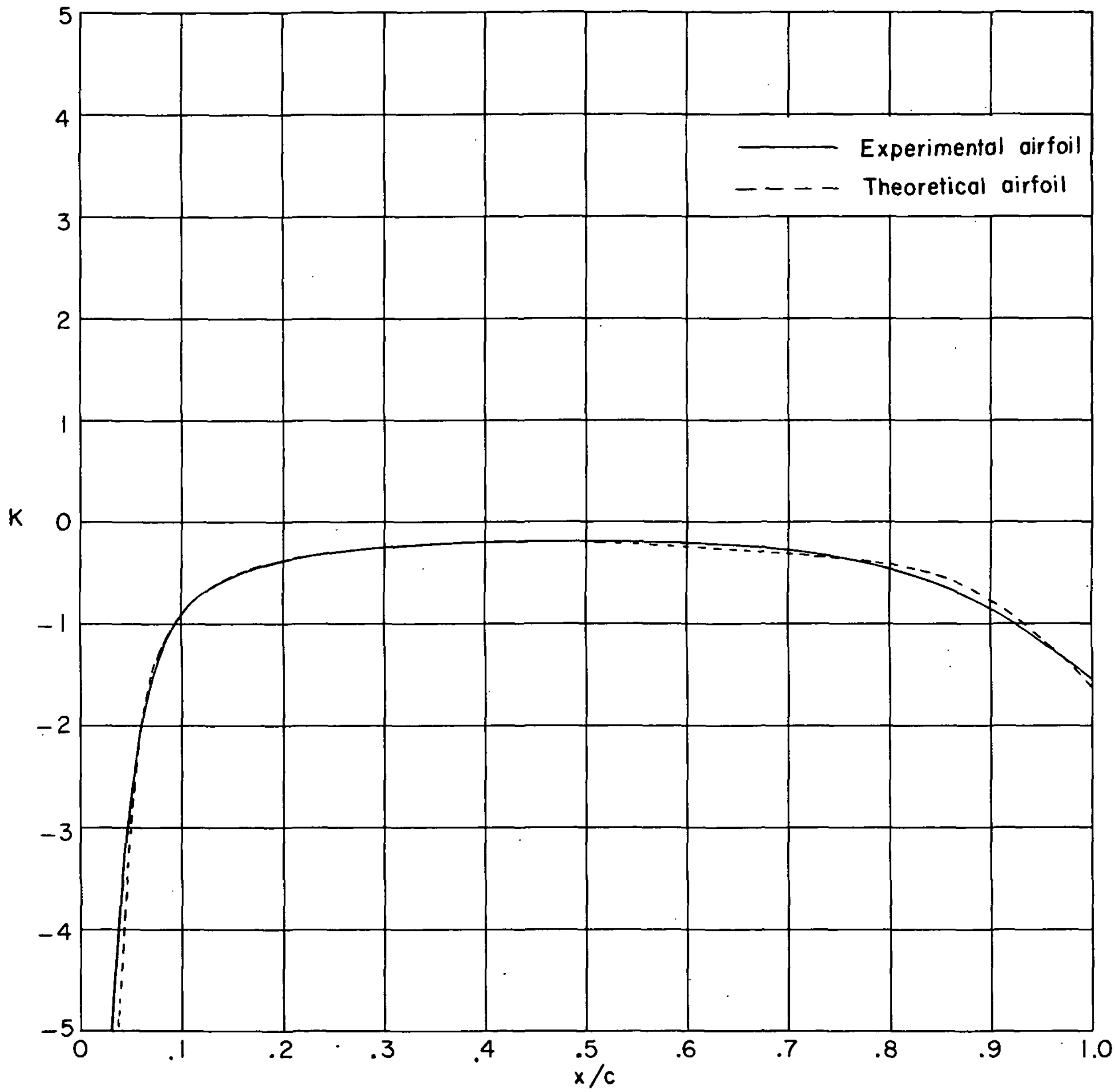


Figure 3.- Upper surface curvature distribution.

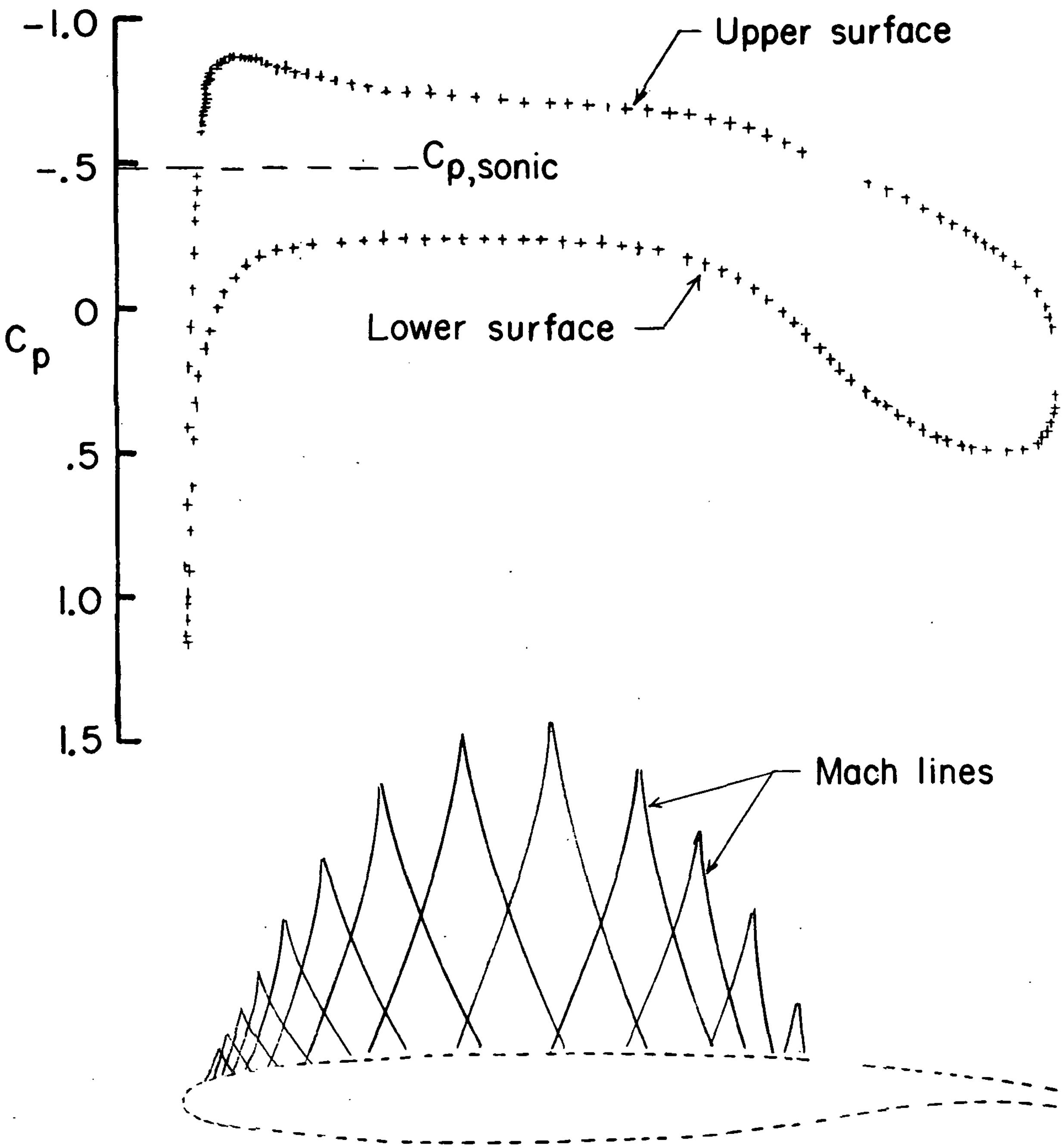
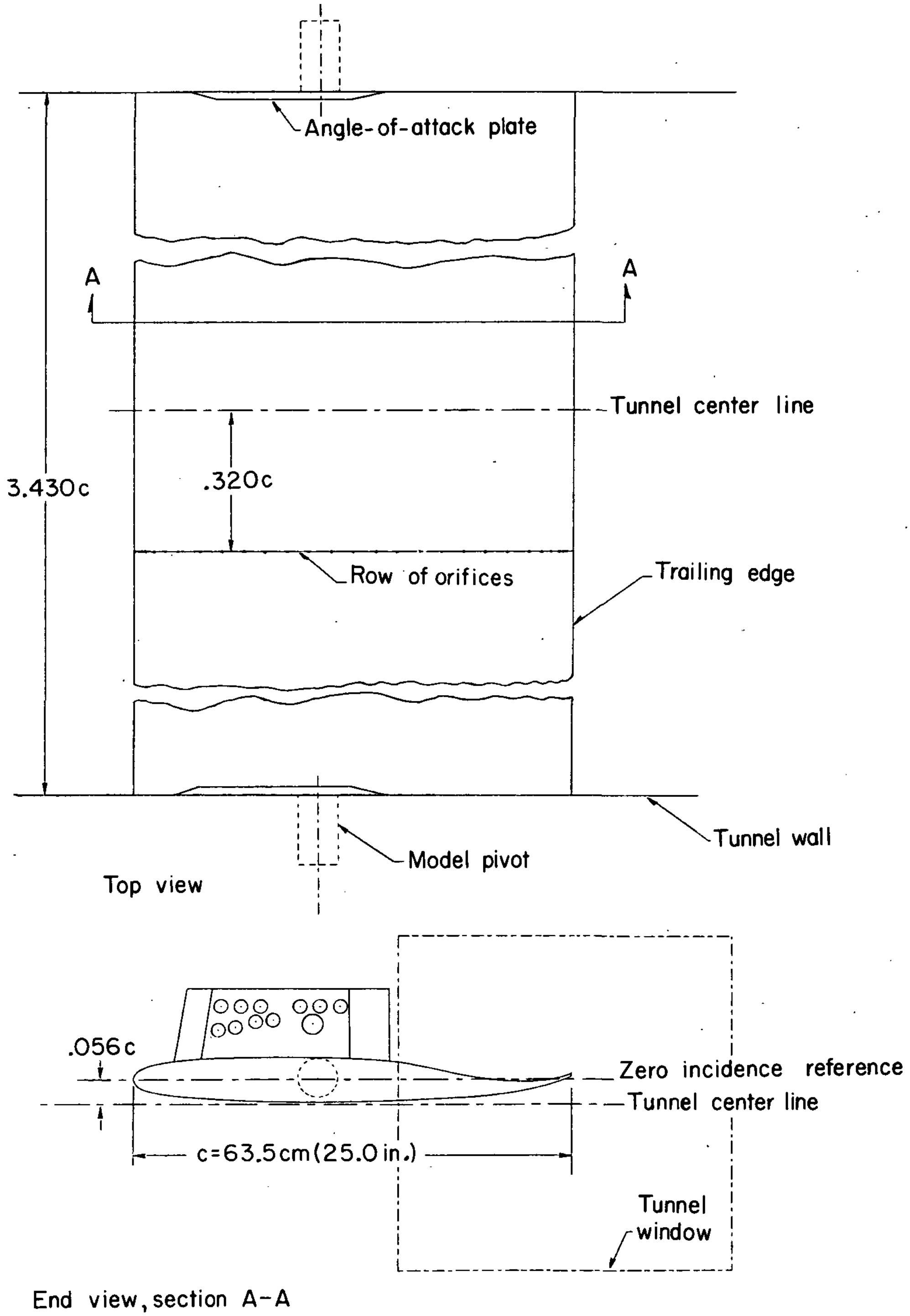
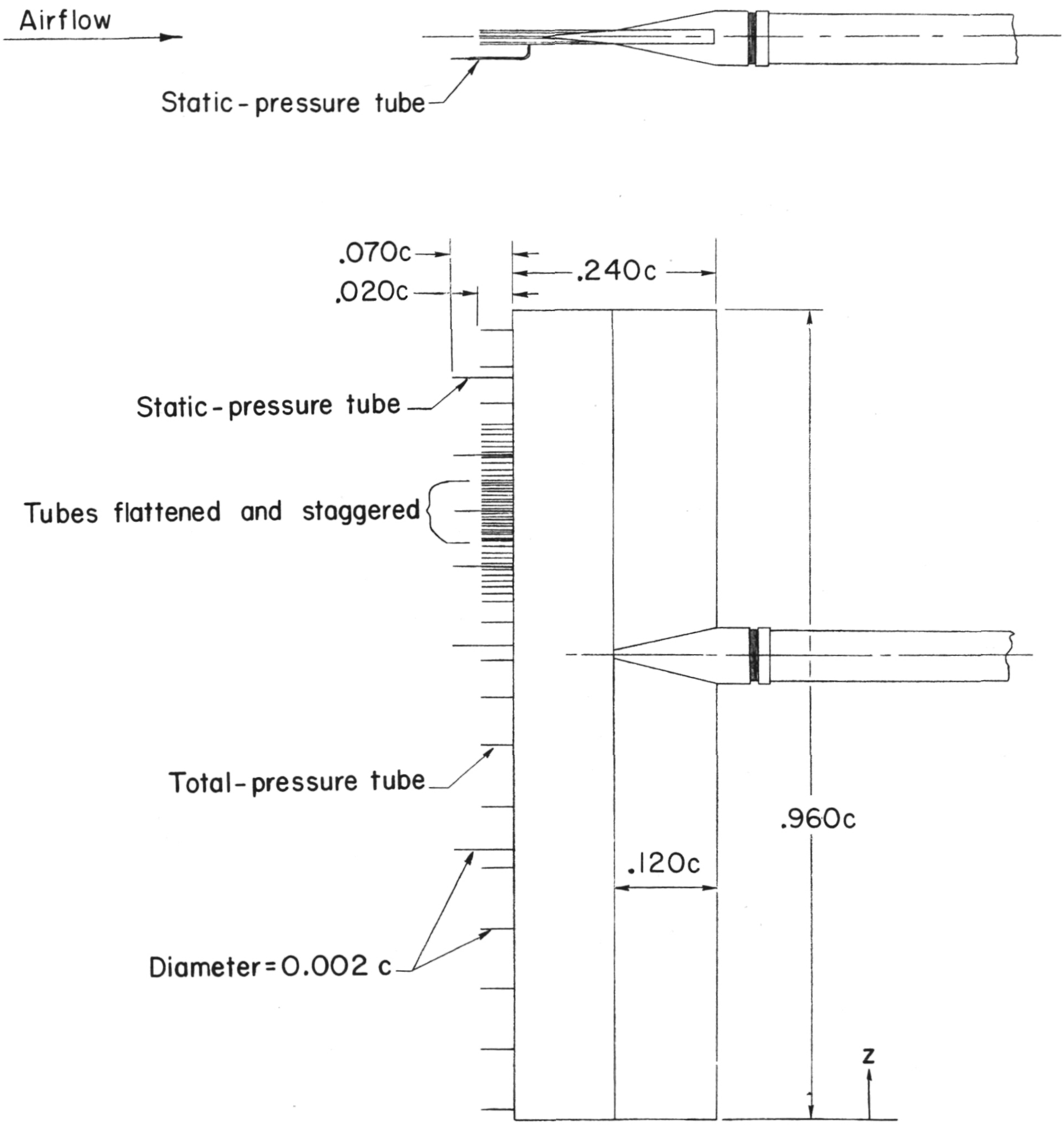


Figure 4.- Theoretical inviscid shockless lifting airfoil with indicated Mach lines in supersonic zone. M = 0.78; c_n = 0.59.



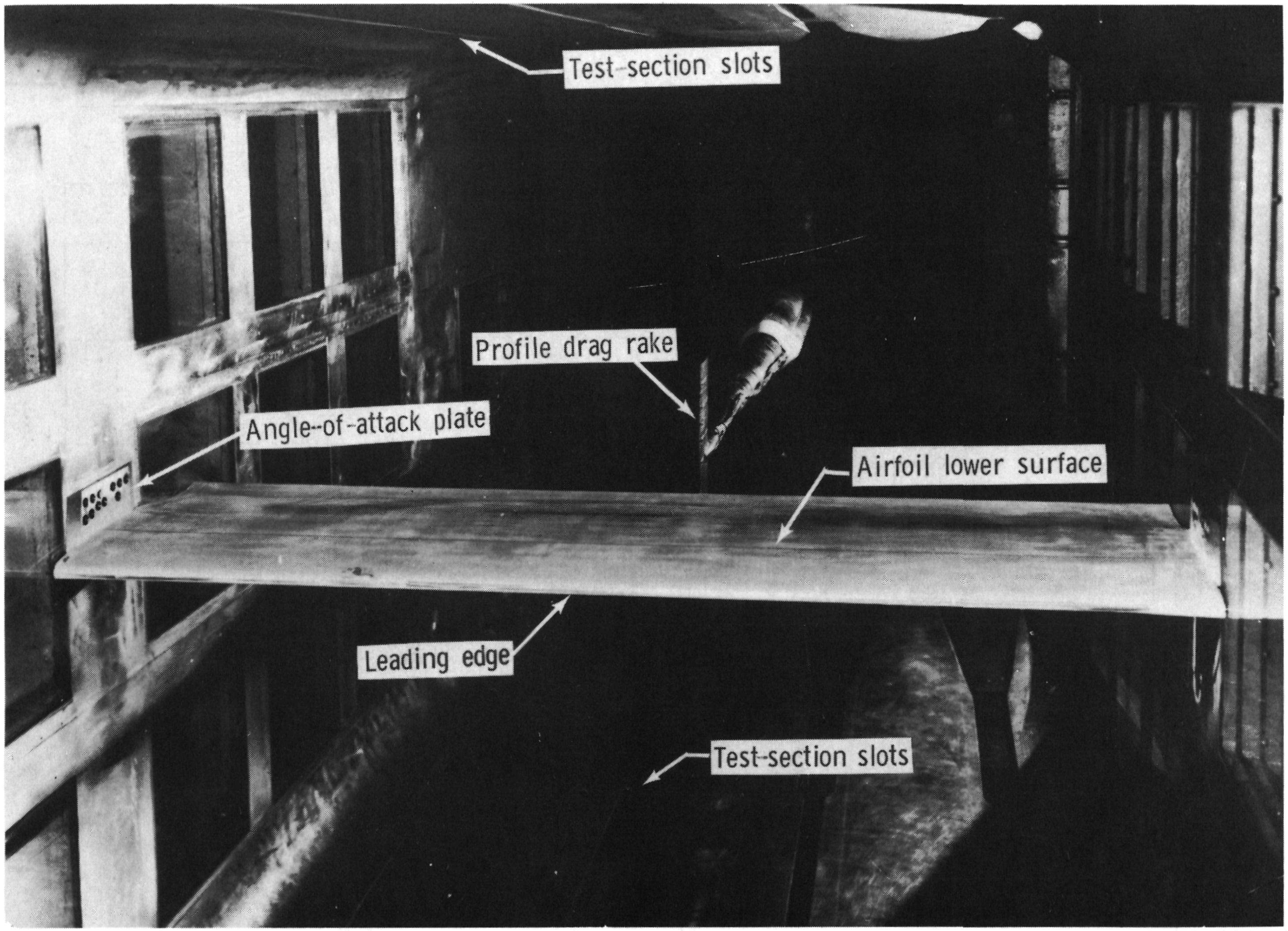
(a) Airfoil mounted in tunnel.

Figure 5.- Apparatus.



(b) Profile drag rake.

Figure 5.- Concluded.

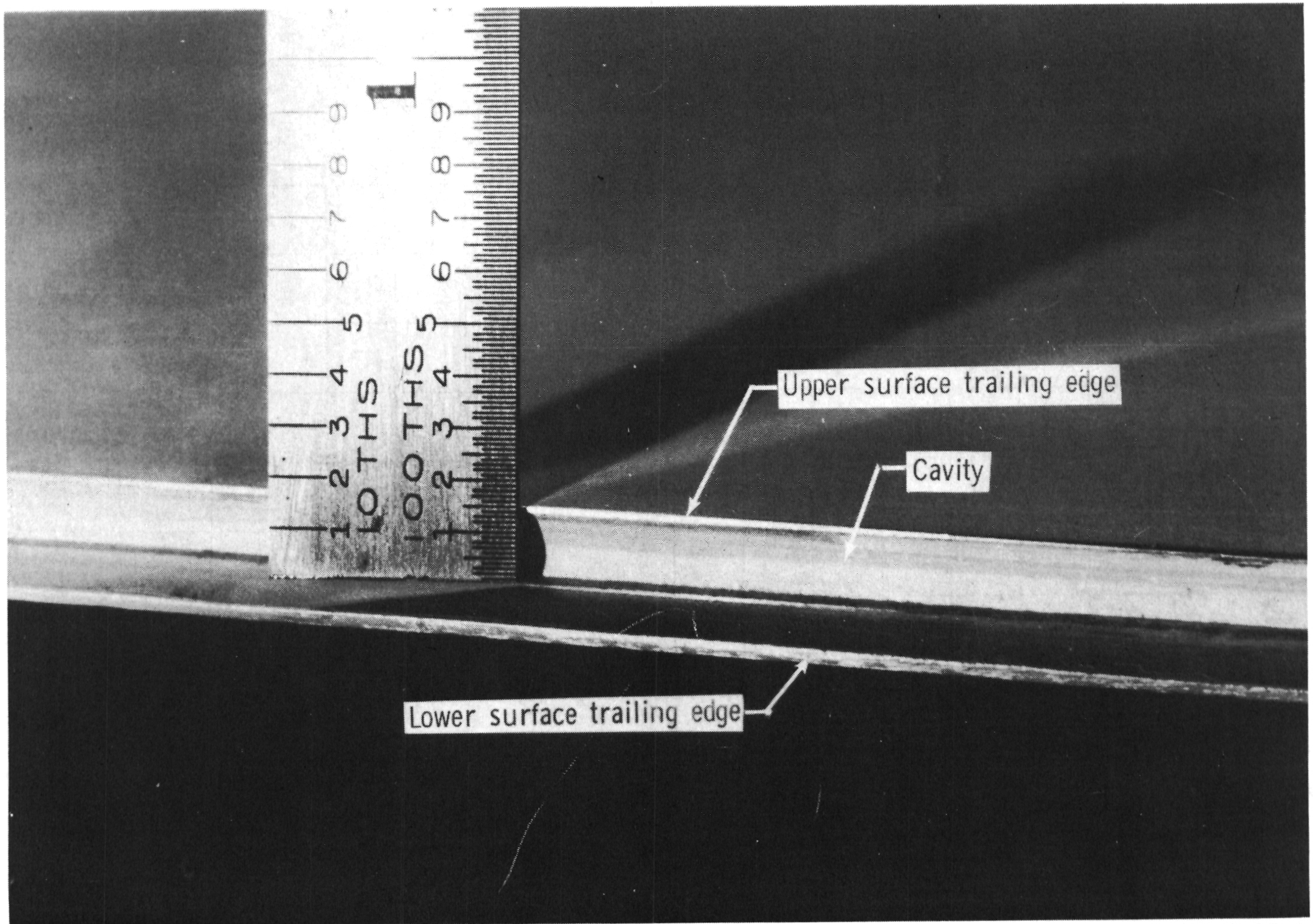


L-73-1225.2

(a) Supercritical airfoil and profile drag rake mounted in tunnel.

Figure 6.- Photographs.

[redacted]



L-73-1227.1

(b) Trailing-edge cavity.

Figure 6.- Concluded.

[redacted]

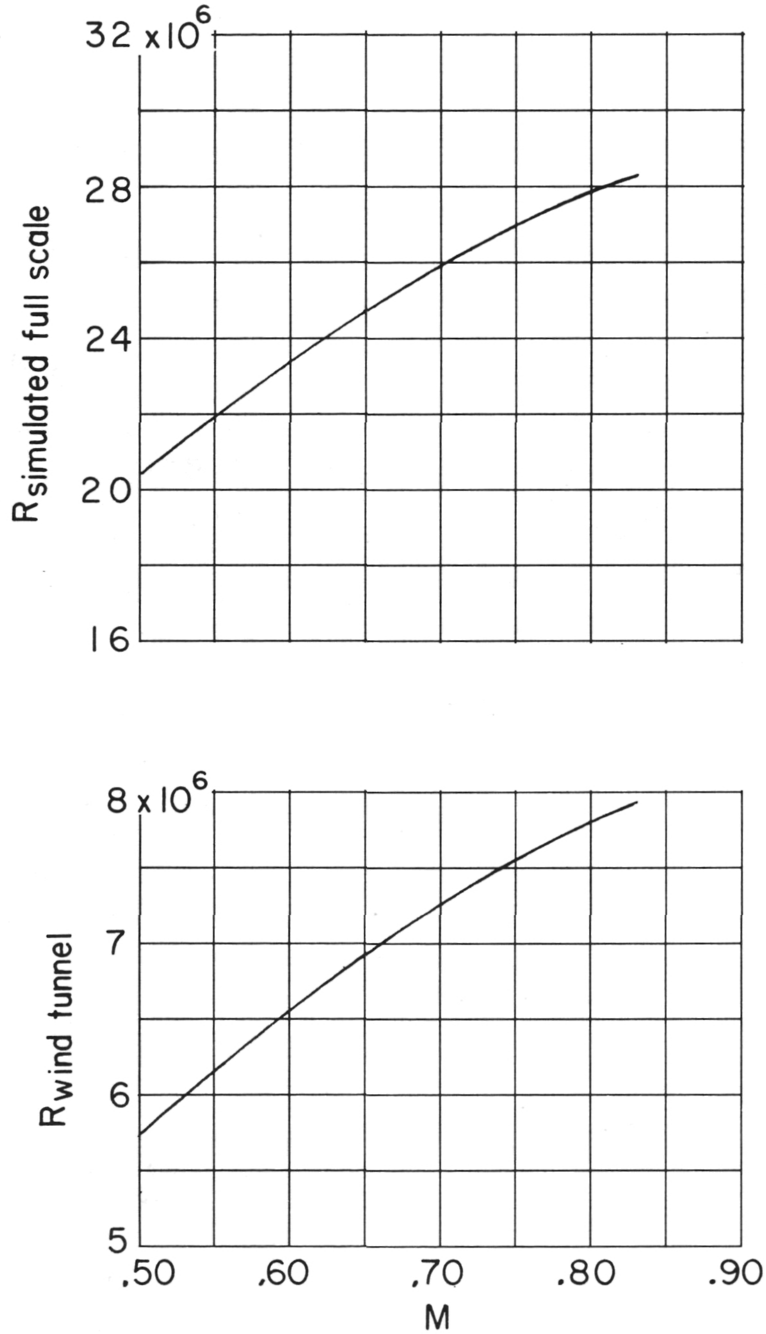
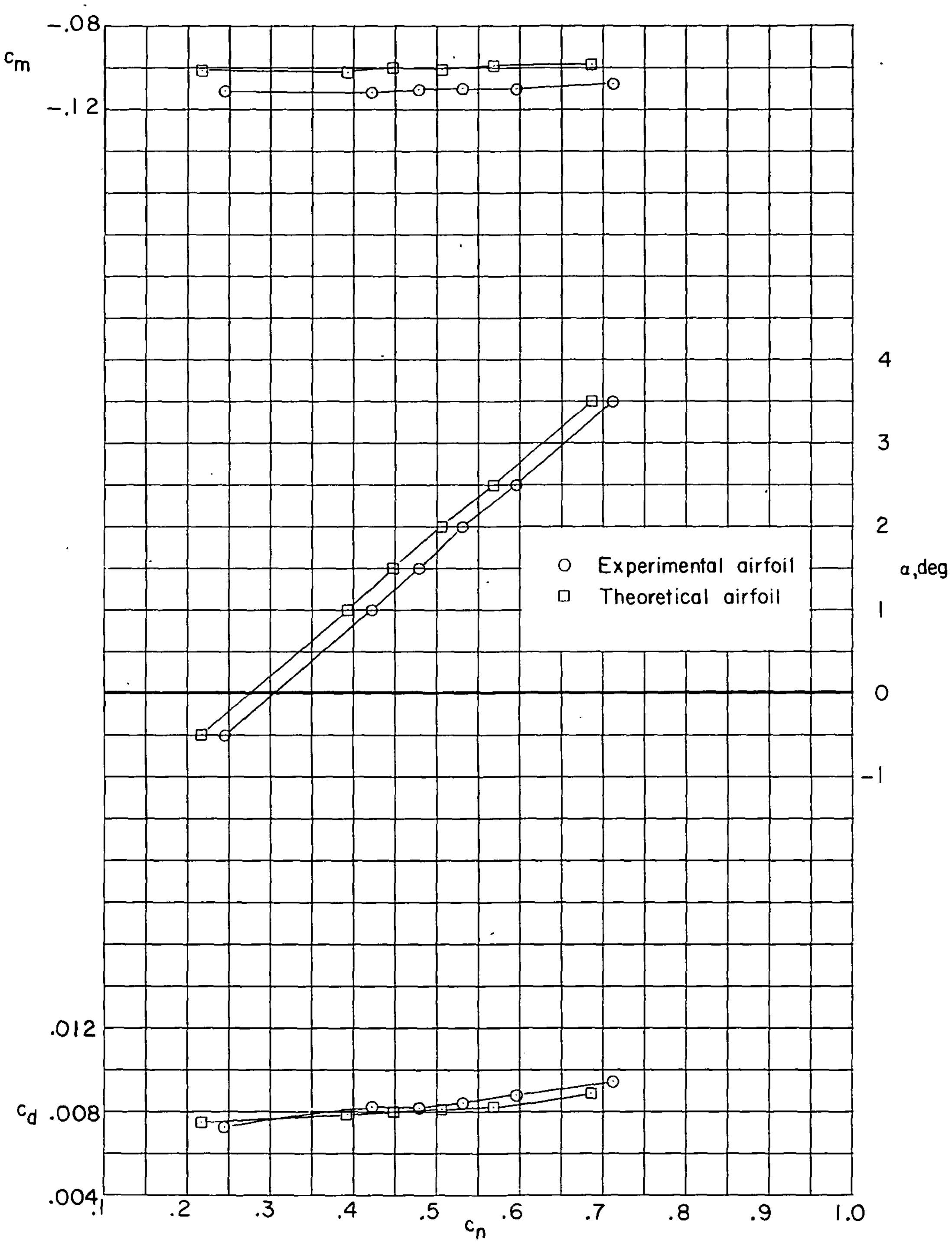
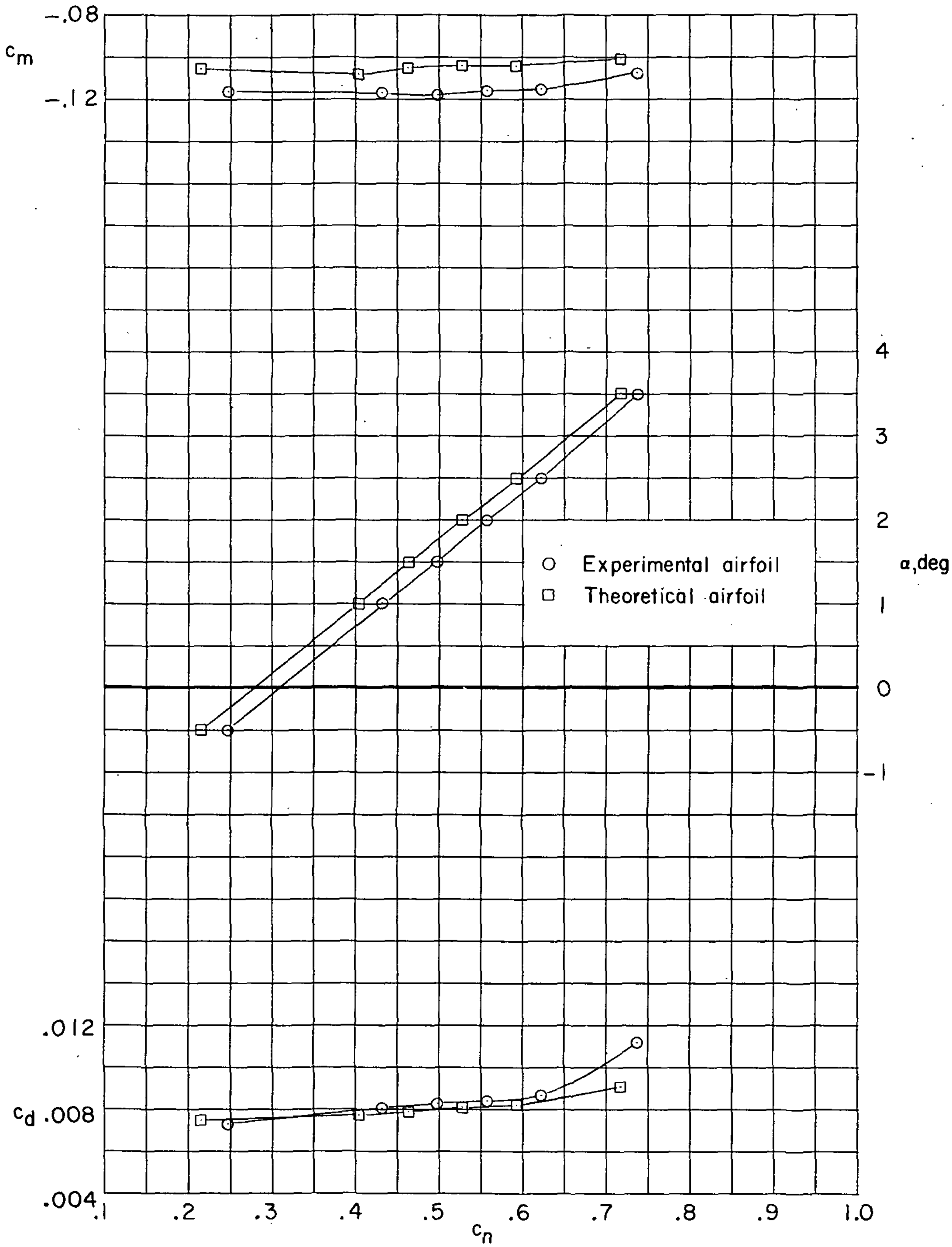


Figure 7.- Variation of test Reynolds number with Mach number.



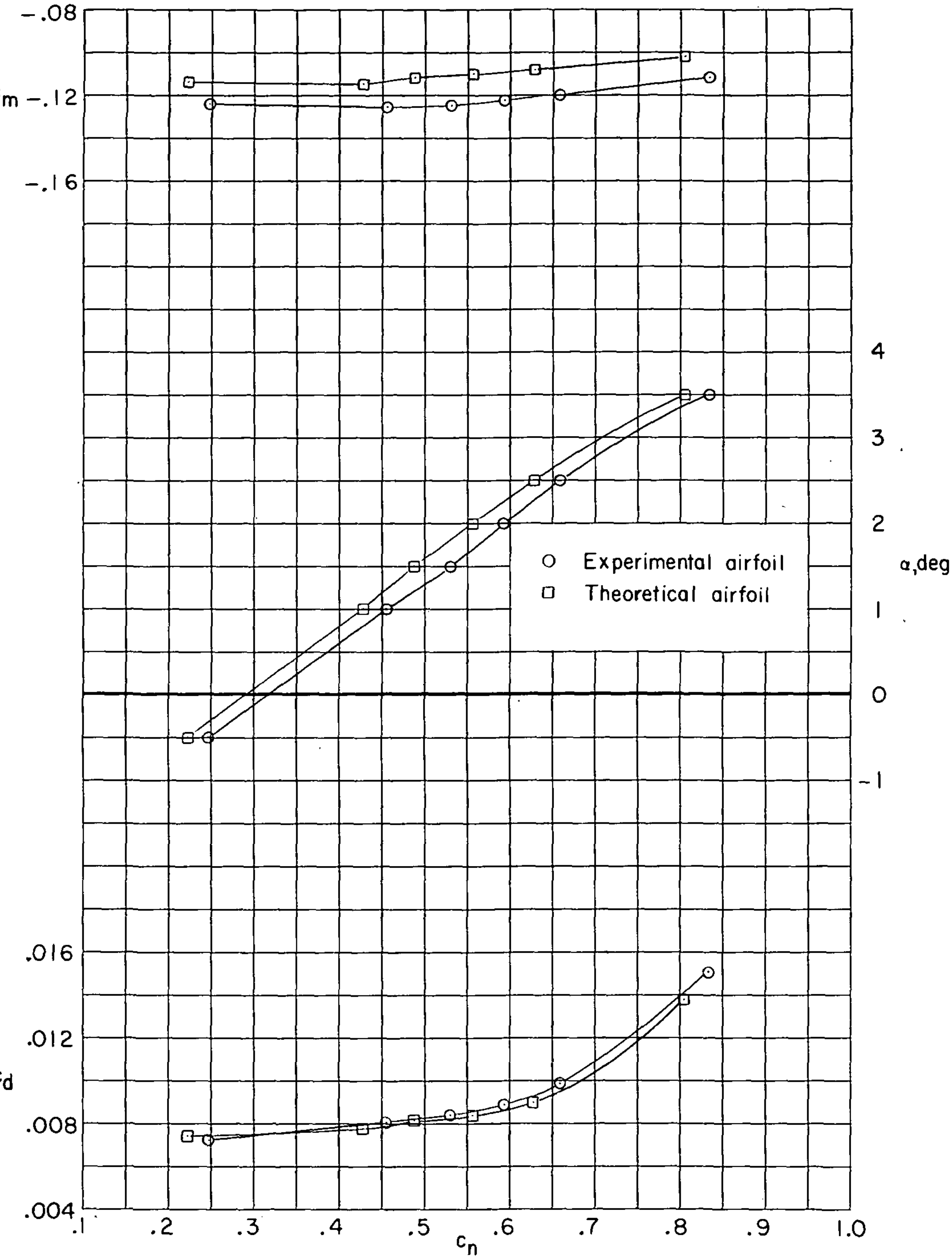
(a) M = 0.50.

Figure 8.- Comparison of force and moment characteristics of experimental and theoretical supercritical airfoils.



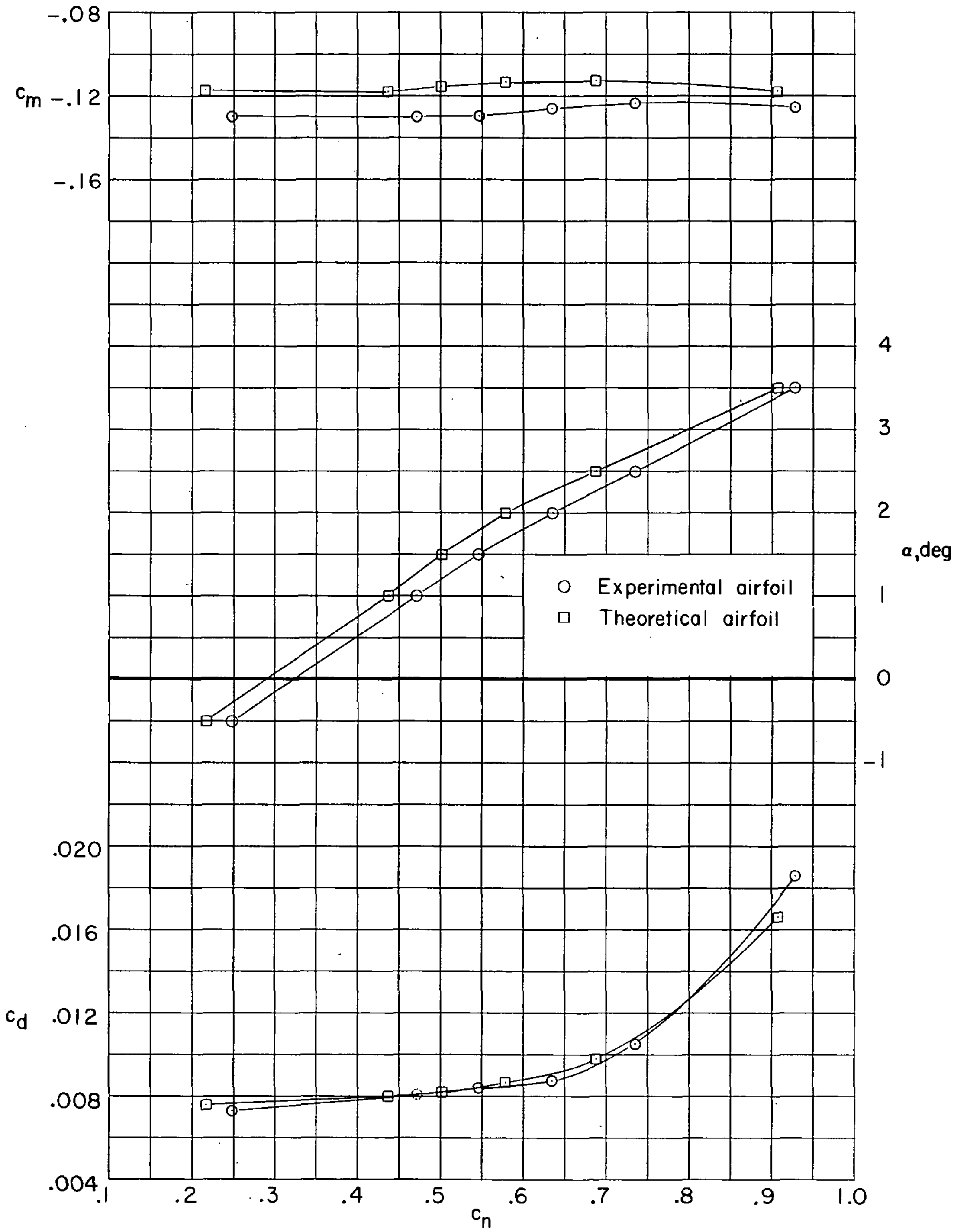
(b) M = 0.60.

Figure 8.- Continued.



(c) M = 0.70.

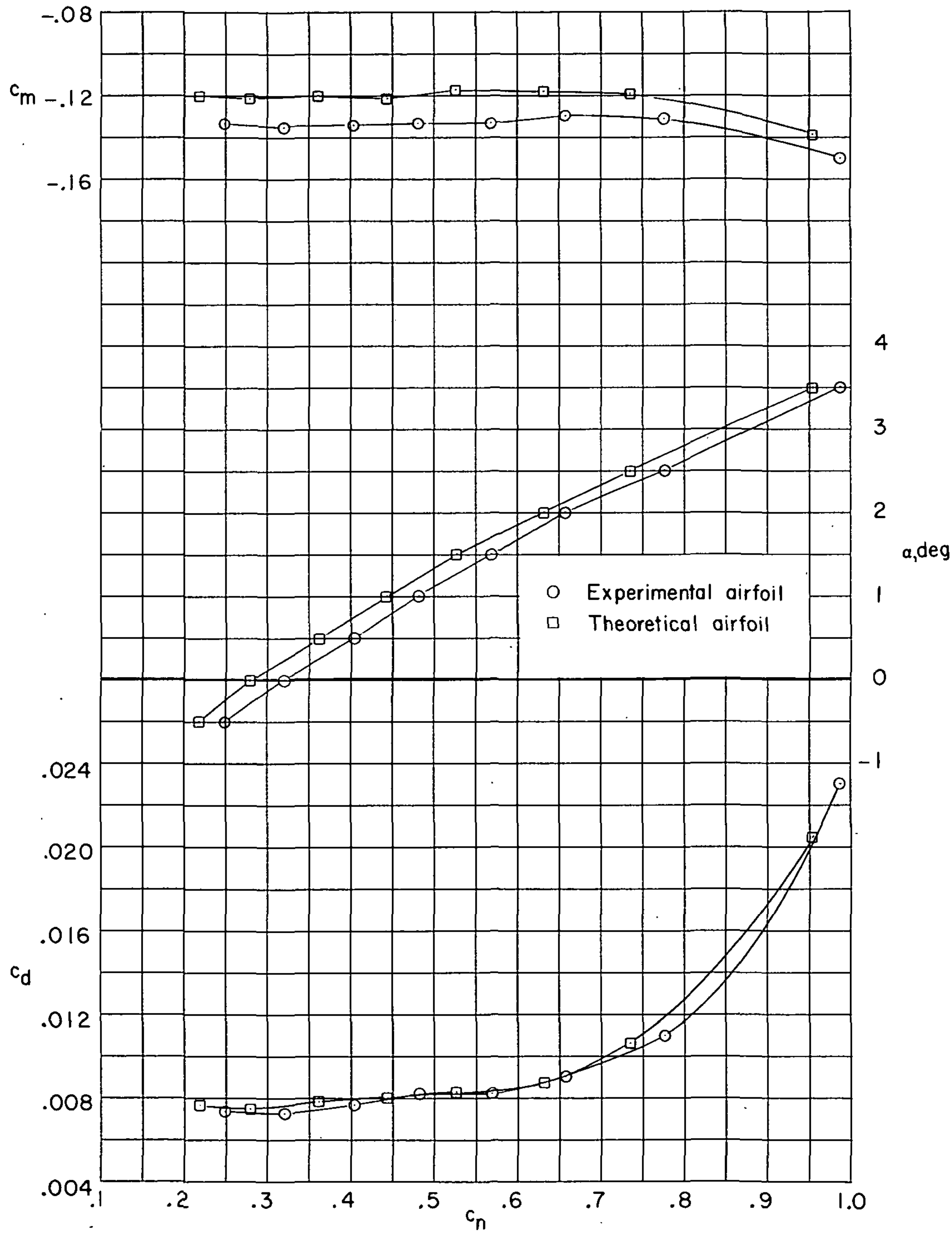
Figure 8.- Continued.



(d) M = 0.74.

Figure 8.- Continued.

[redacted]

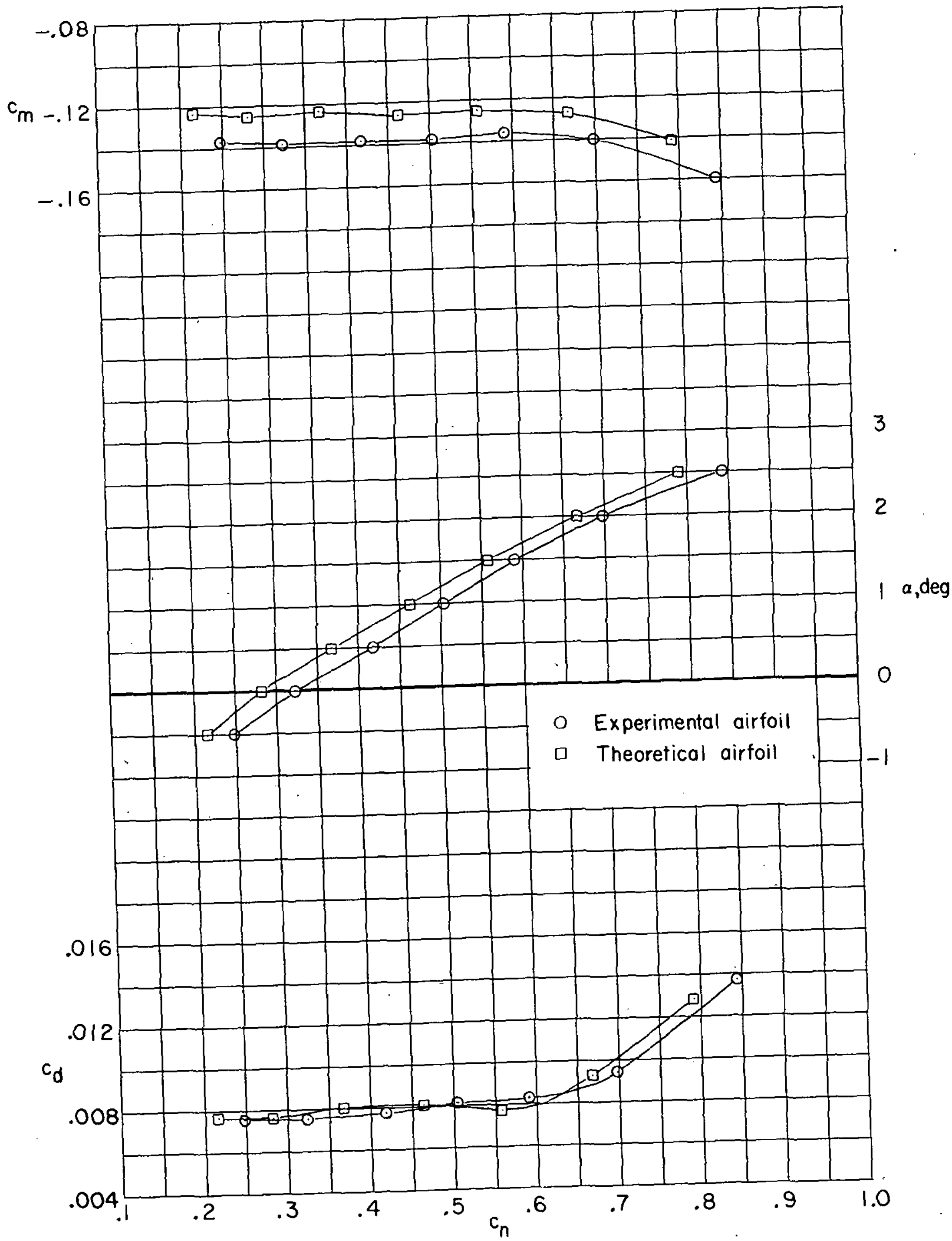


(e) M = 0.76.

Figure 8.- Continued.

[redacted]

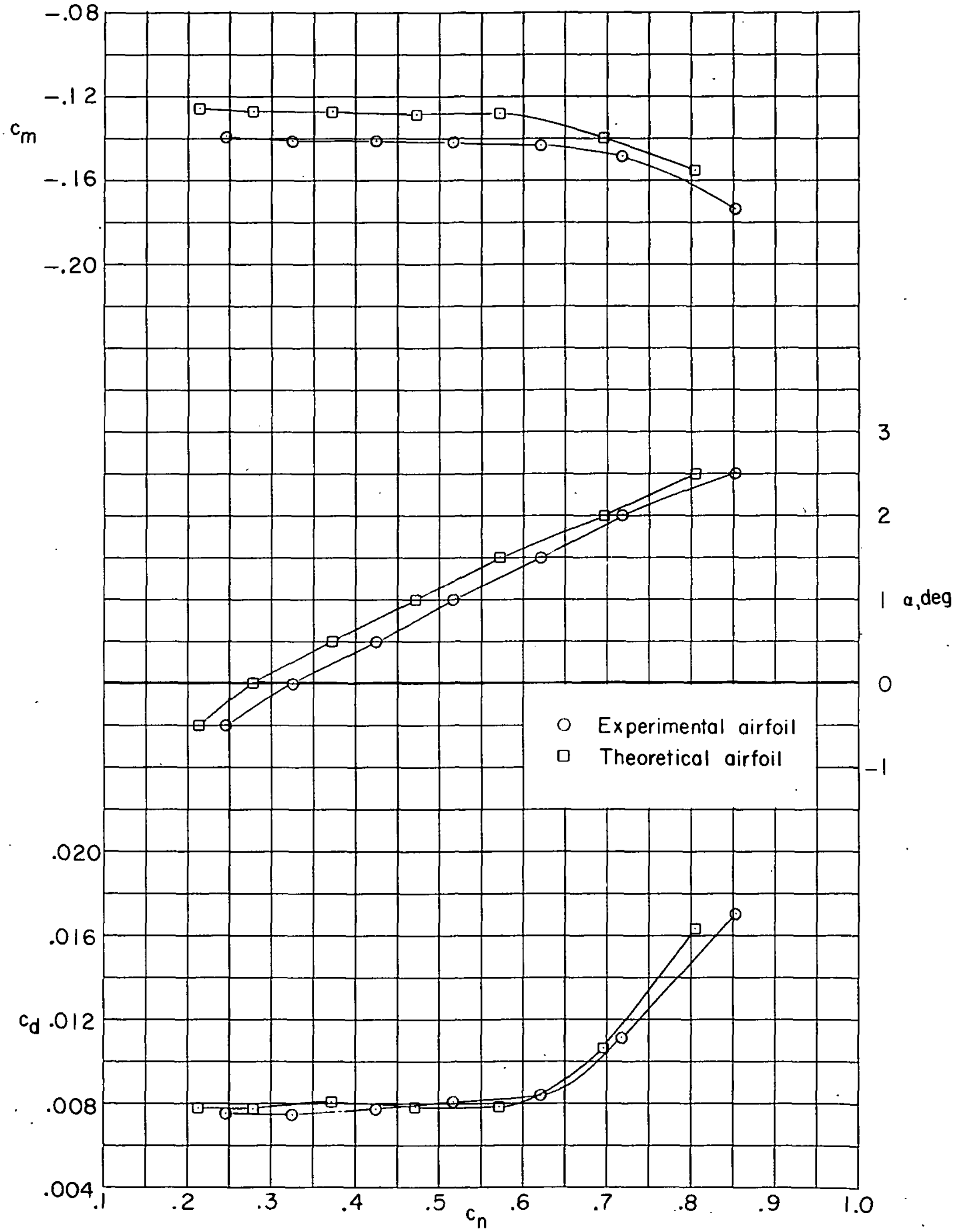
[redacted]



(f) M = 0.78.

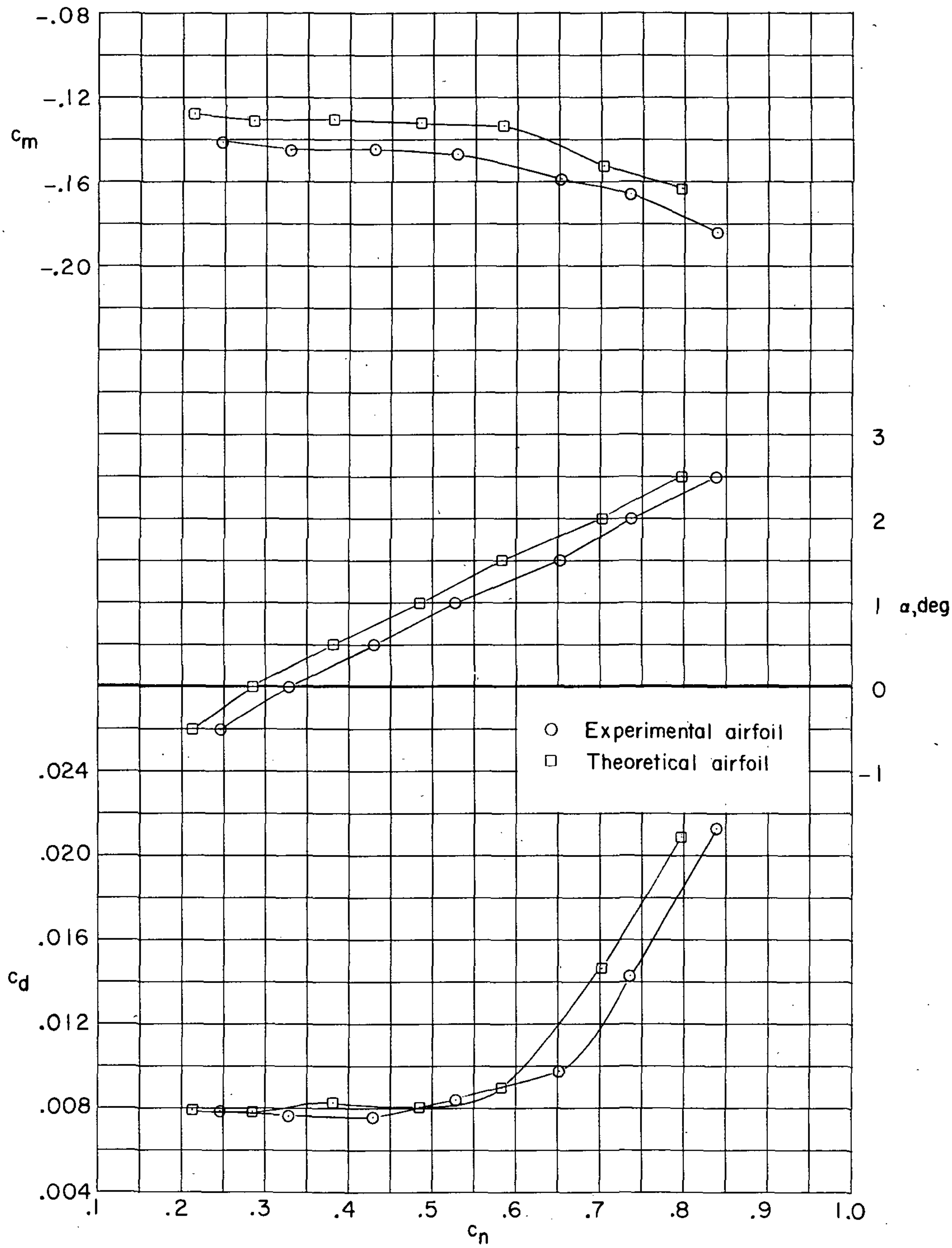
Figure 8.- Continued.

[redacted]



(g) M = 0.79.

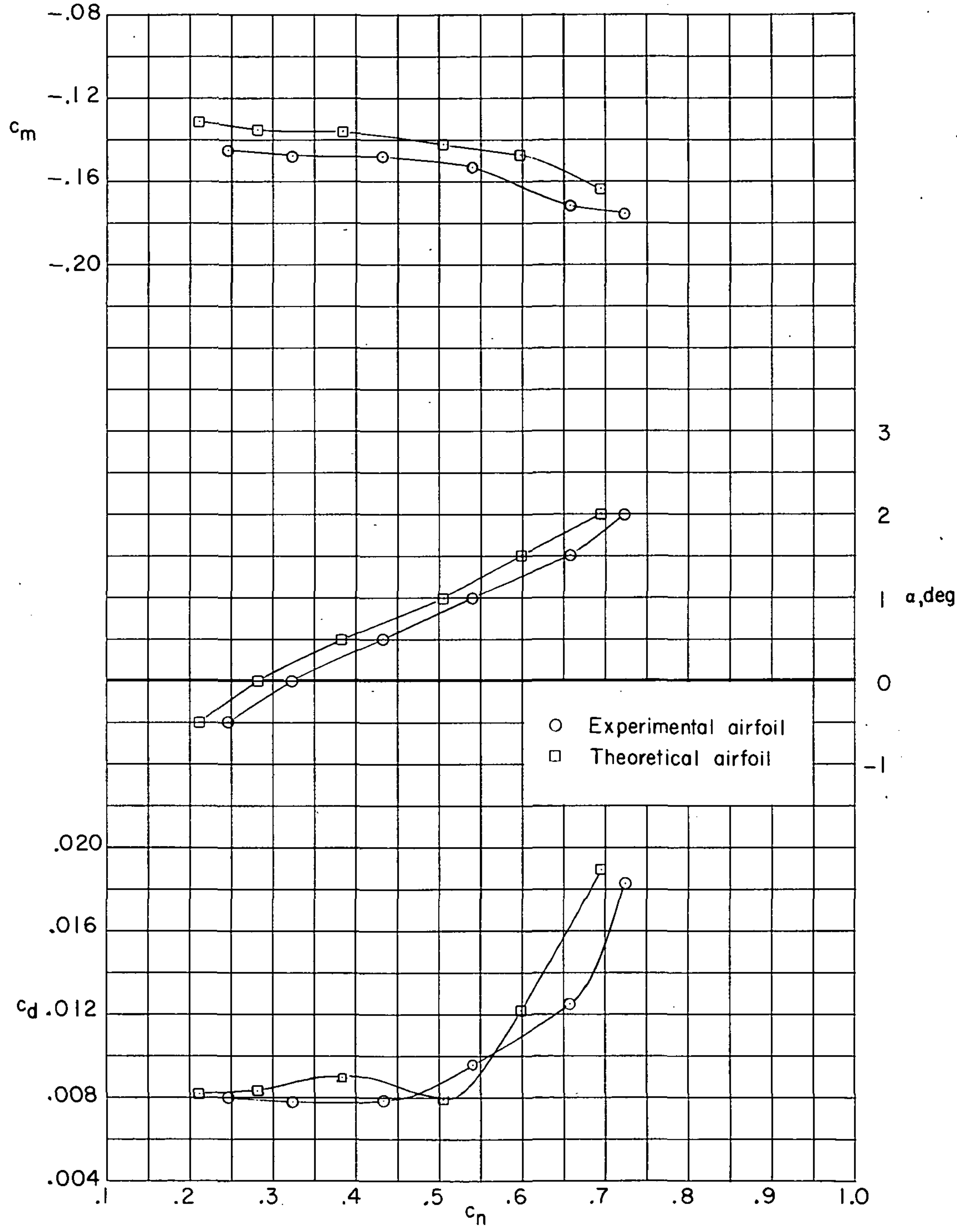
Figure 8.- Continued.



(h) M = 0.80.

Figure 8.- Continued.

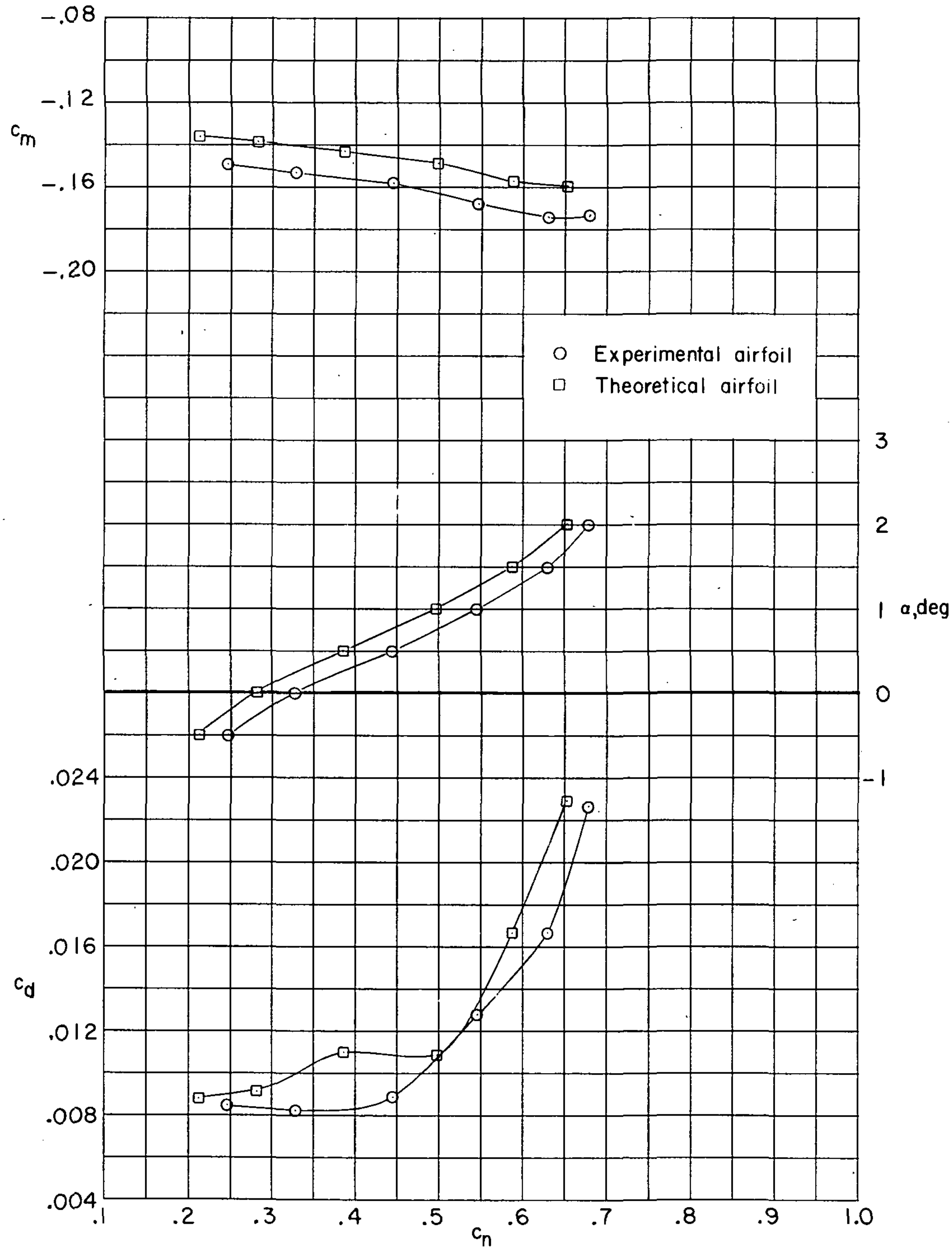
[redacted]



(i) M = 0.81.

Figure 8.- Continued.

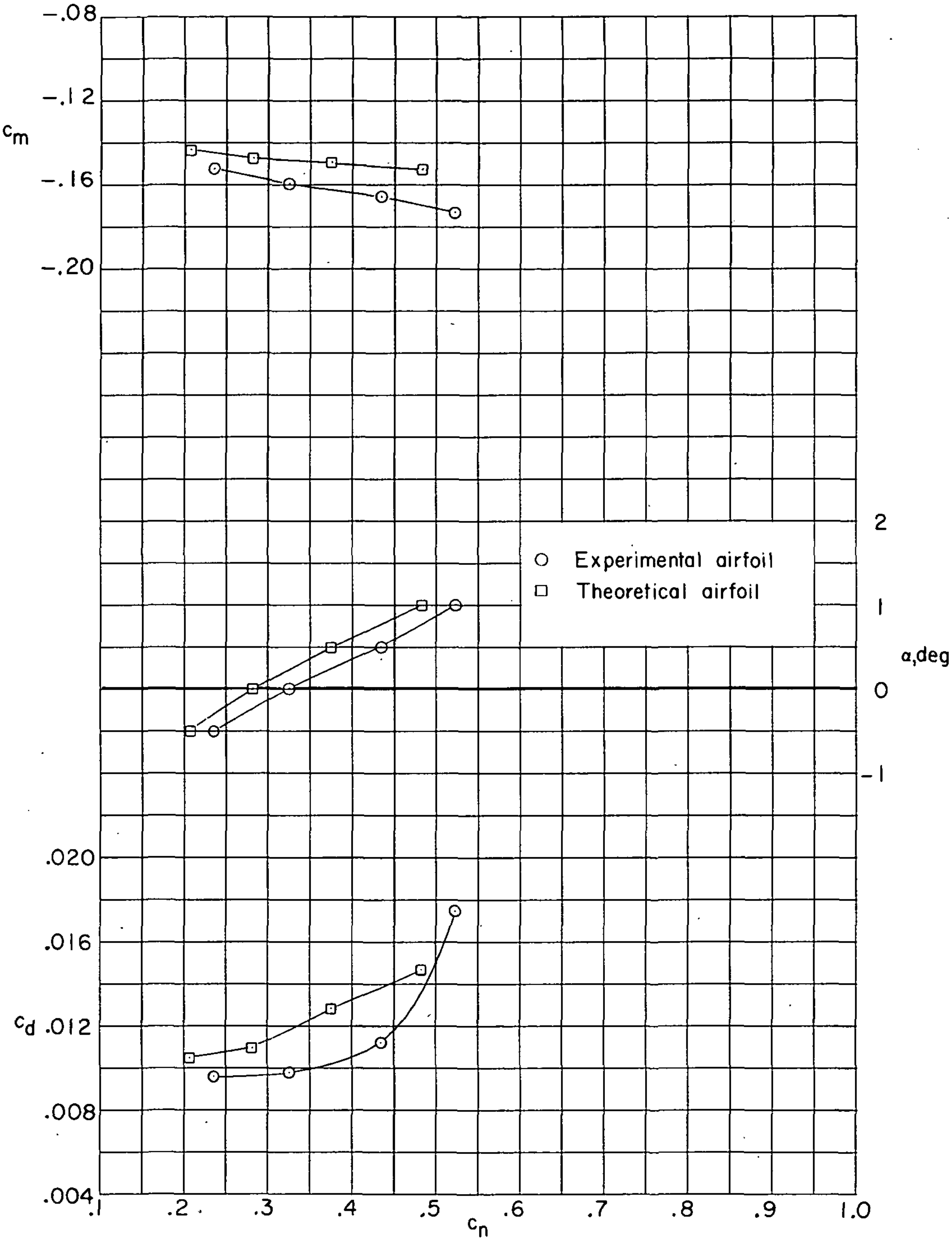
[redacted]



(j) M = 0.82.

Figure 8.- Continued.

[redacted]



(k) M = 0.83.

Figure 8.- Concluded.

[redacted]

[redacted]

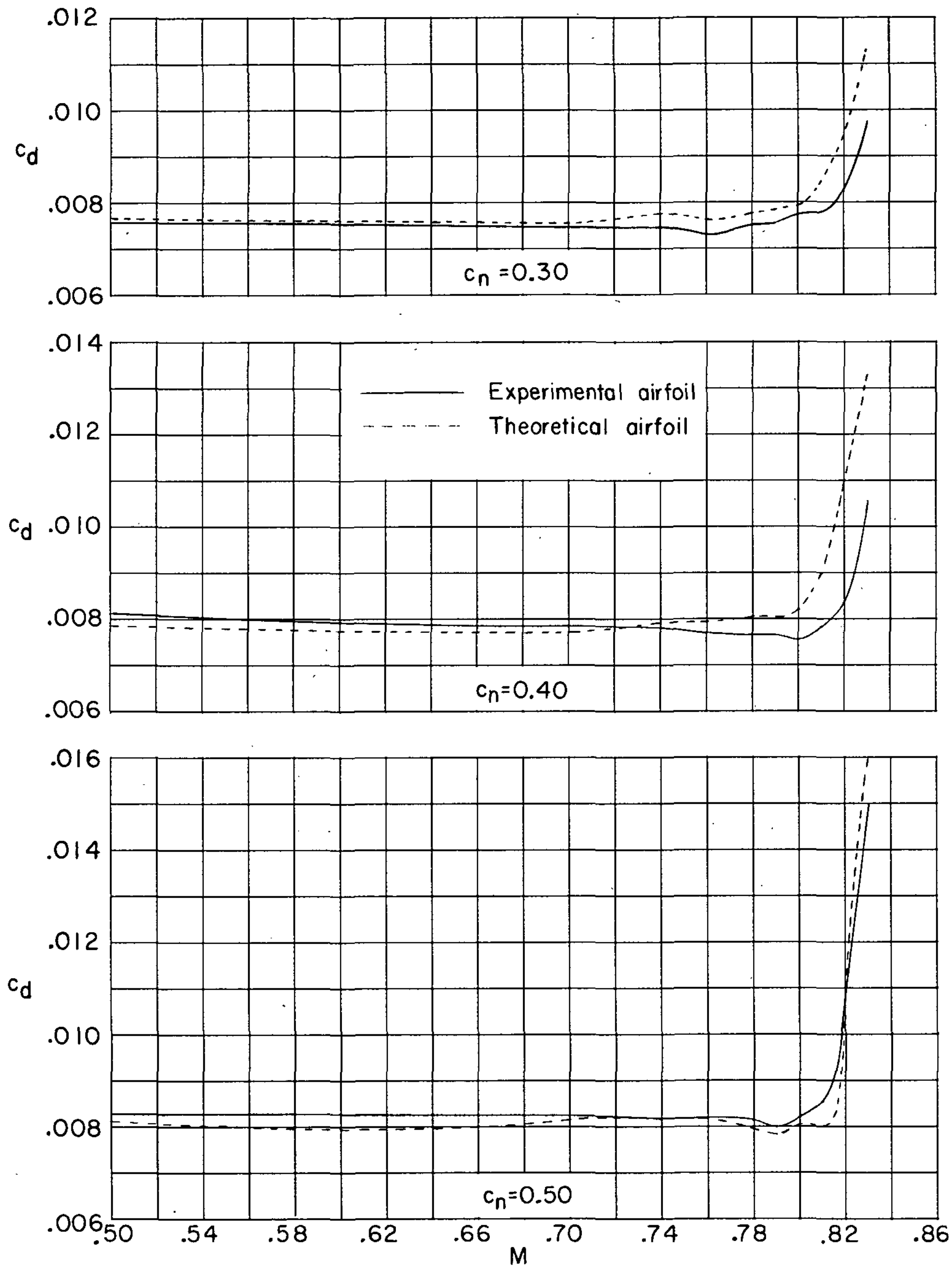


Figure 9.- Variation of section drag coefficient with Mach number of experimental and theoretical supercritical airfoils of various normal-force coefficients.

[redacted]

[redacted]

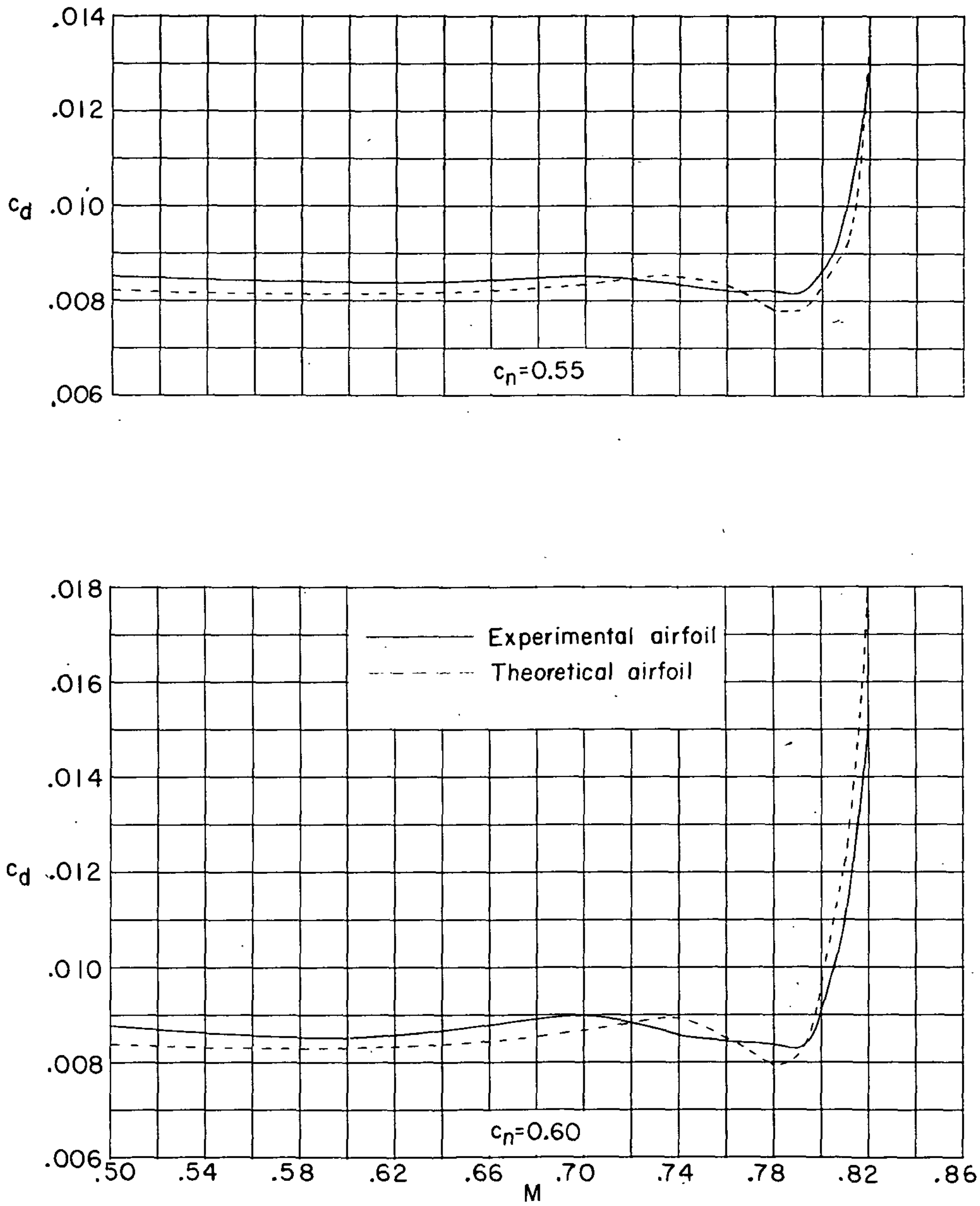
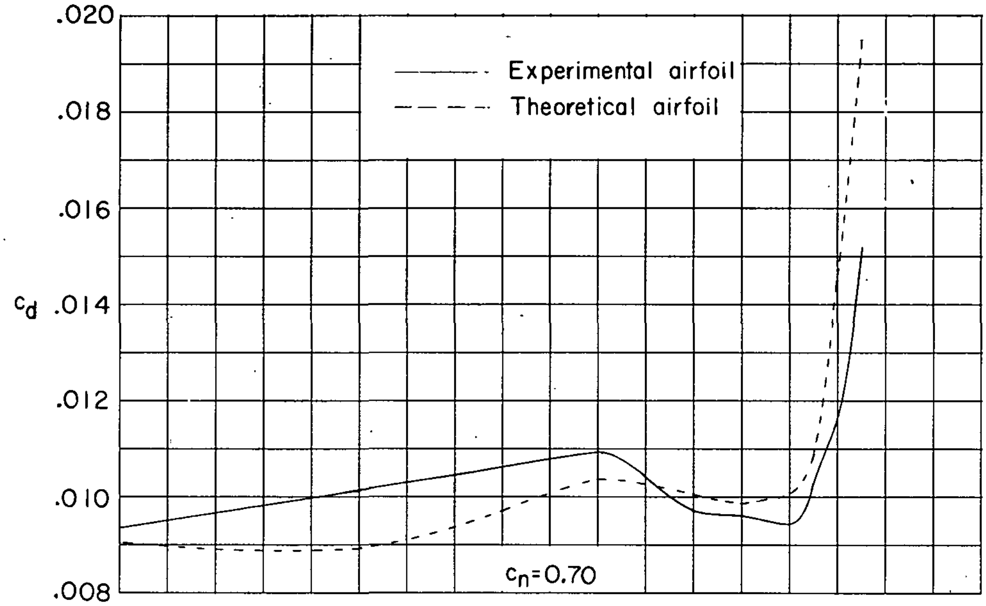


Figure 9.- Continued.

[redacted]

[redacted]



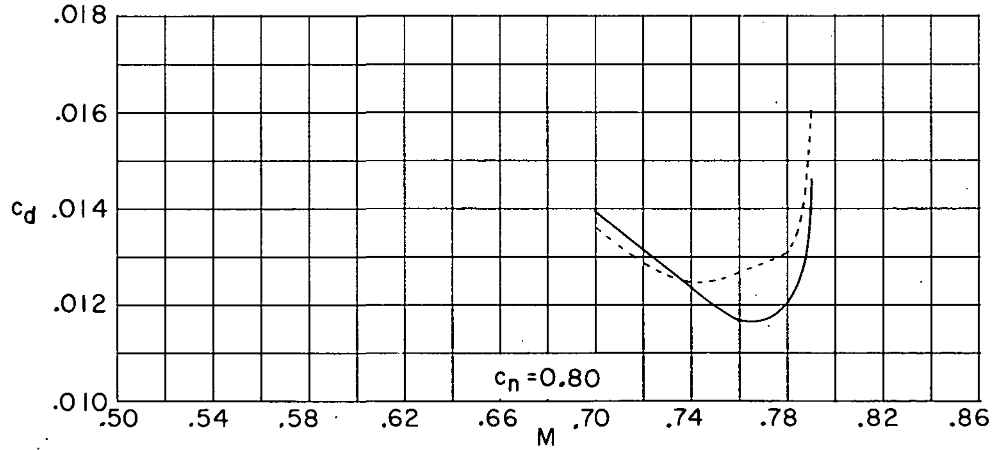


Figure 9.- Concluded.

[redacted]

[redacted]

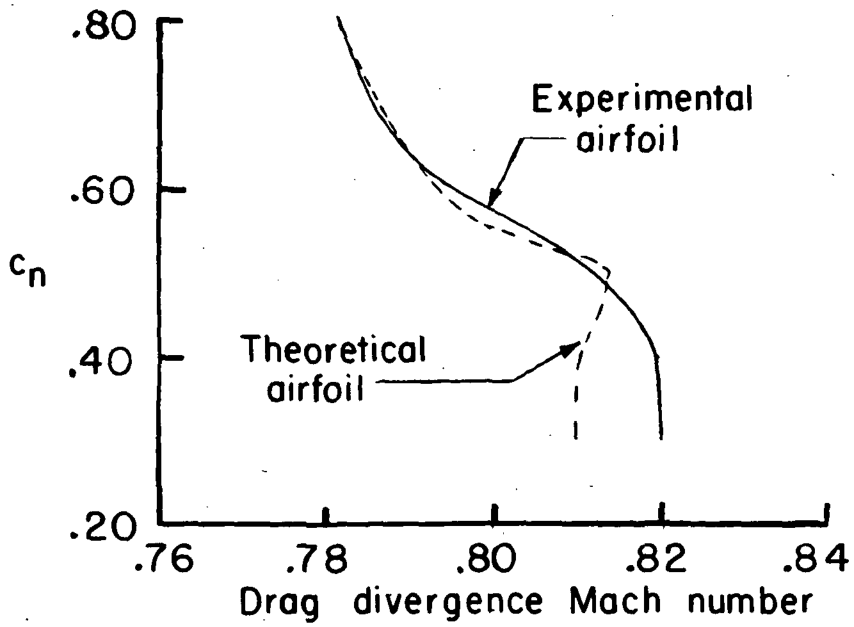


Figure 10.- Drag divergence Mach numbers.

[redacted]

[redacted]

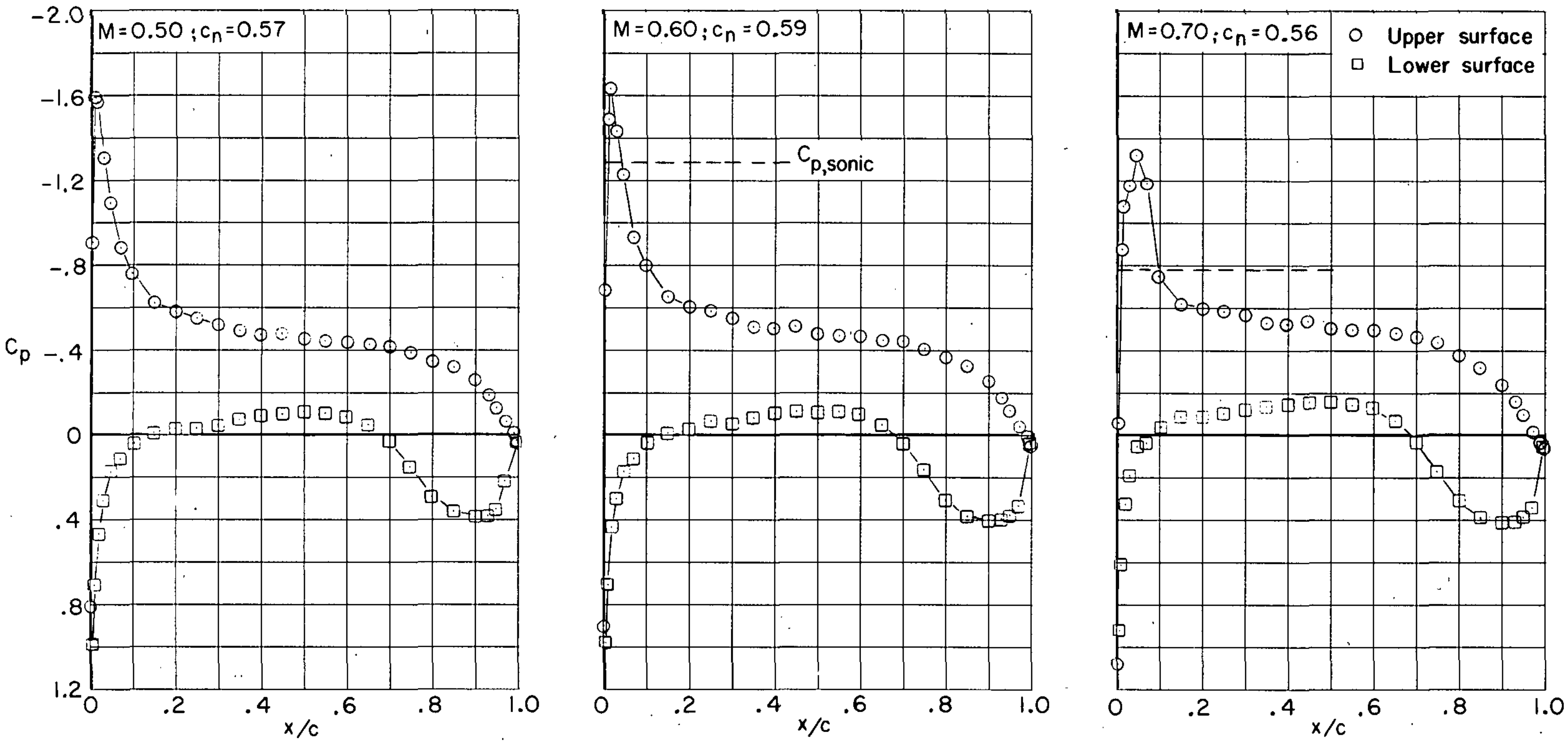


Figure 11.- Representative surface-pressure distributions on theoretical airfoil. c_n near 0.6.

[redacted]

[redacted]

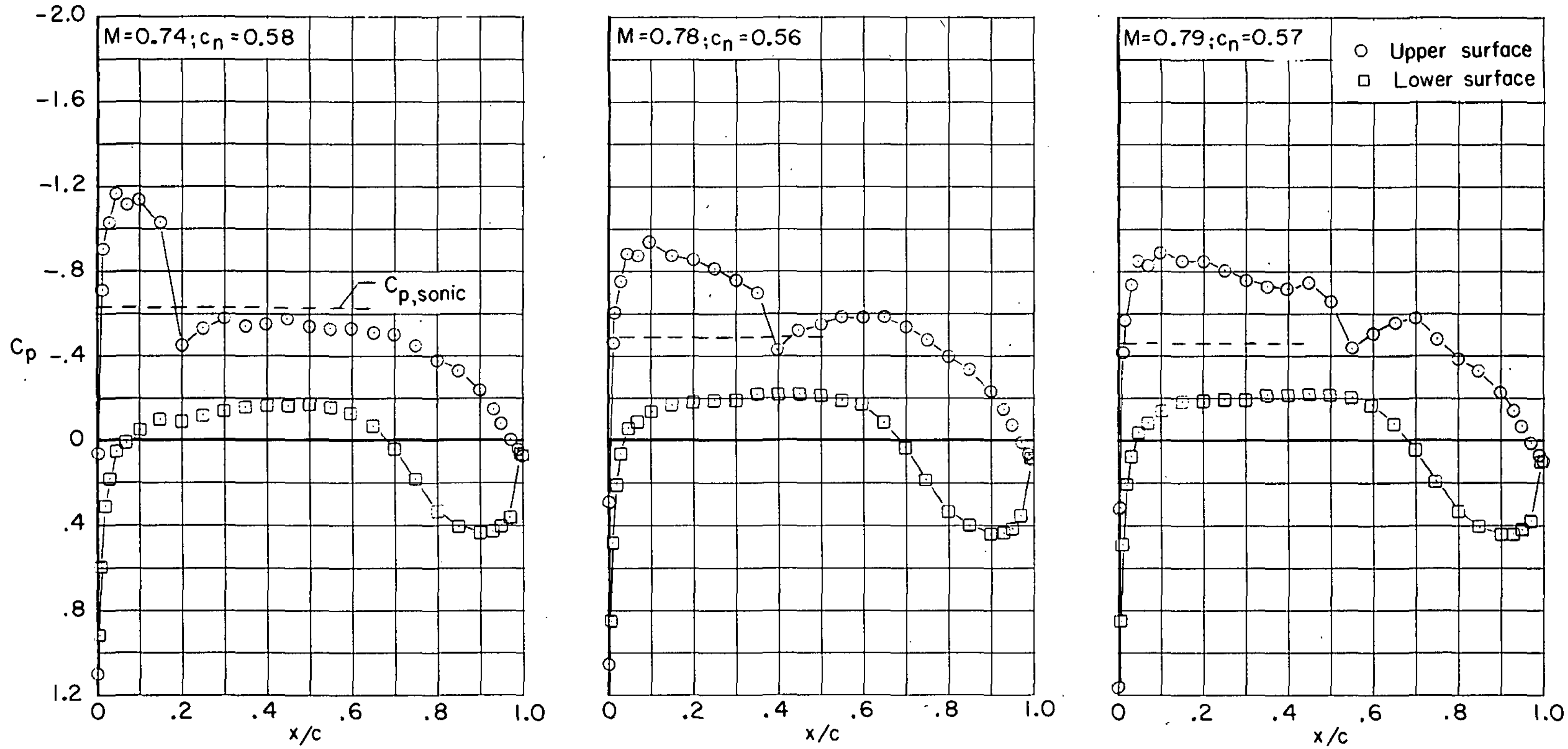


Figure 11.- Concluded.

[redacted]

[redacted]

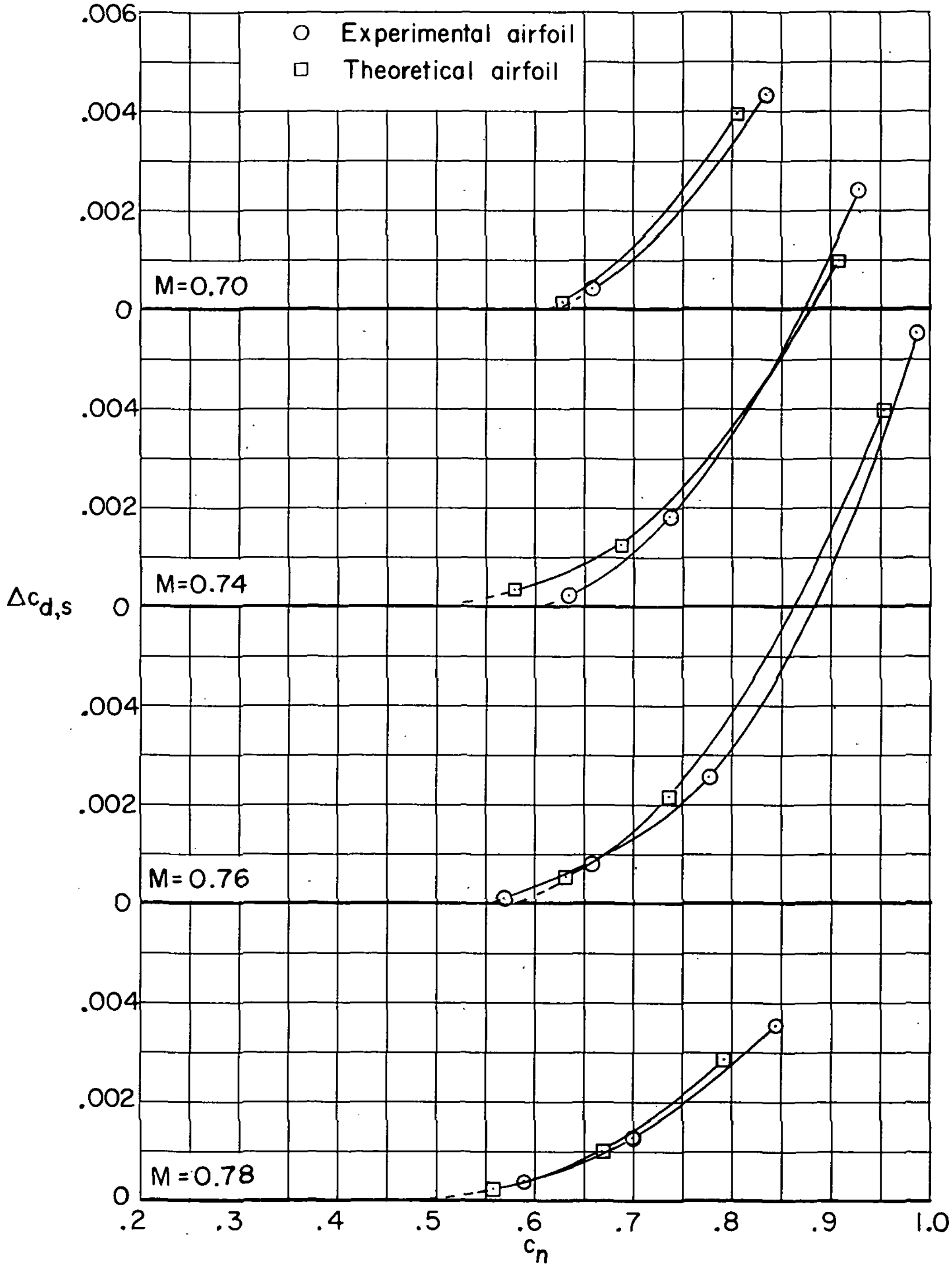


Figure 12.- Drag due to shock loss.

[redacted]

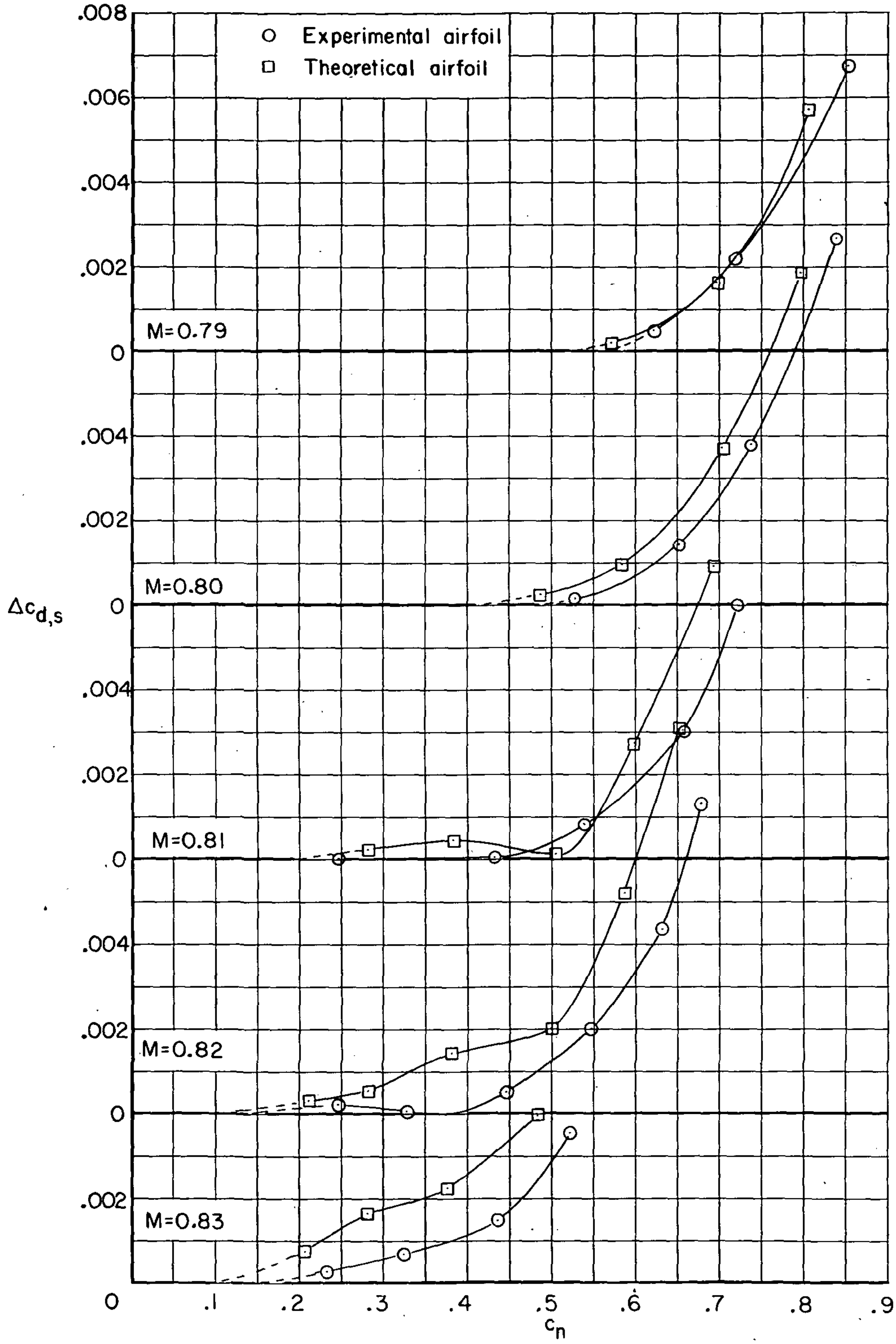


Figure 12.- Concluded.

[redacted]

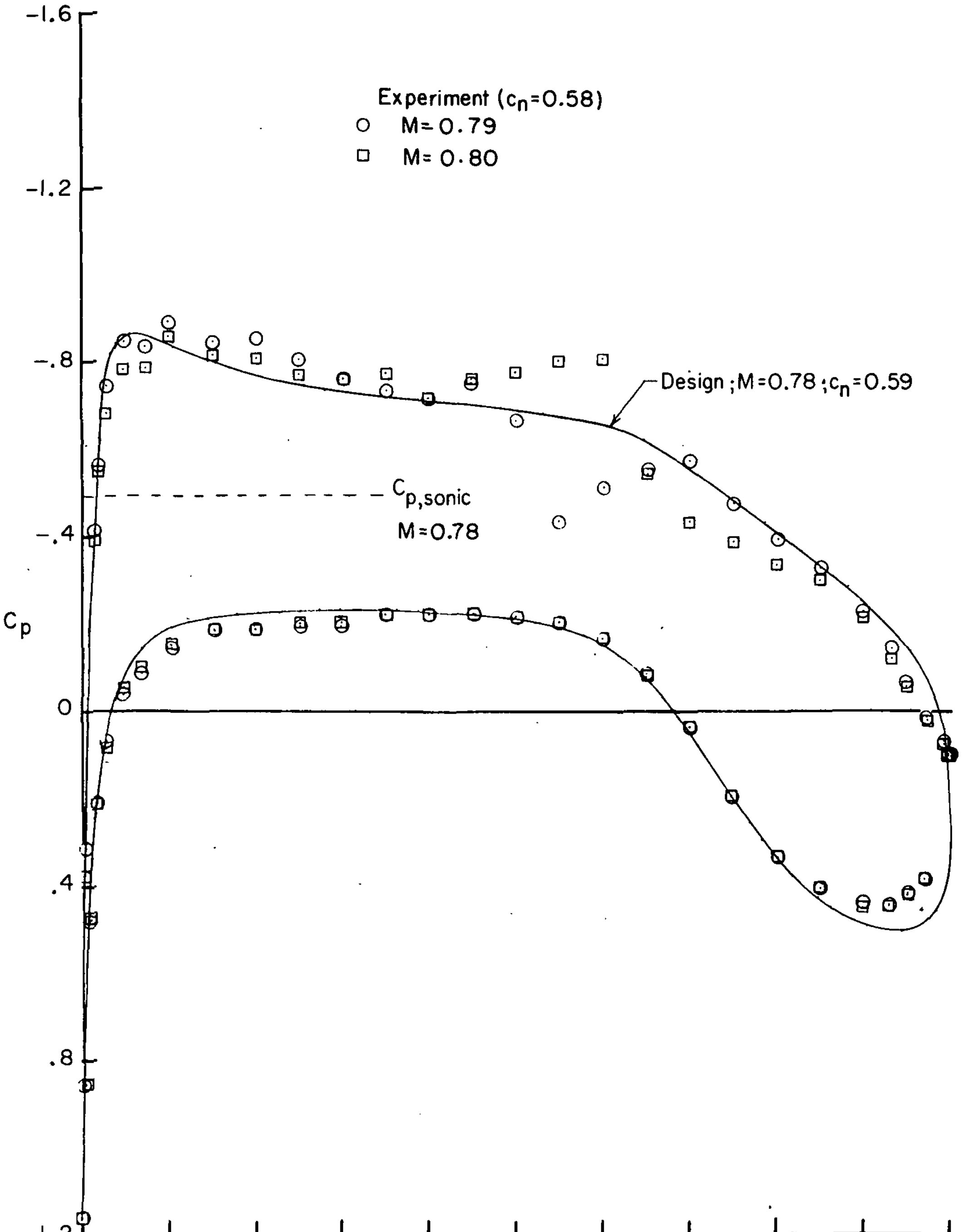


Figure 13.- Comparison of design and experimental pressure distributions of theoretical airfoil.

[redacted]

[redacted]

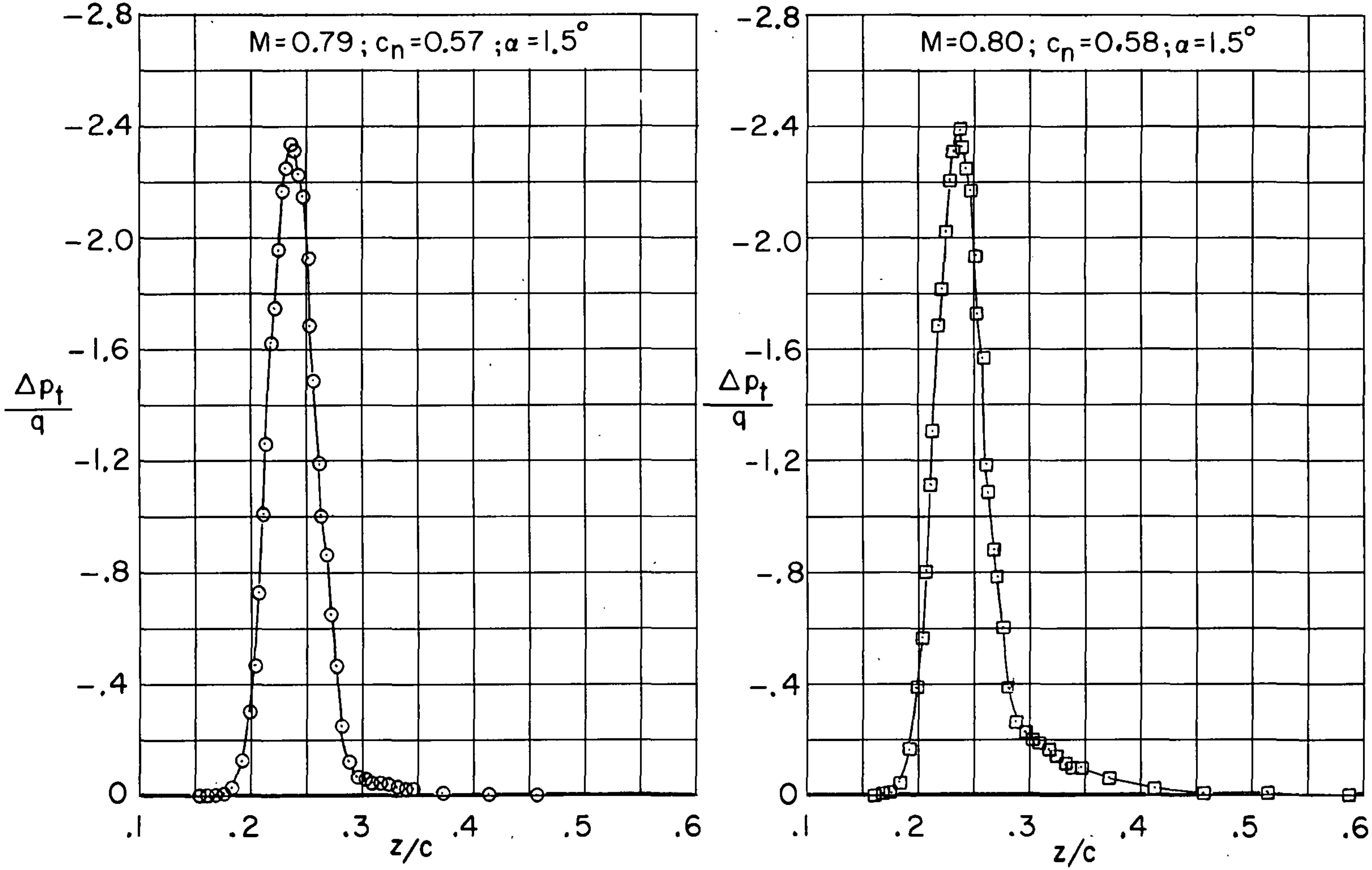


Figure 14.- Representative experimental wake profiles for theoretical airfoil.

[redacted]

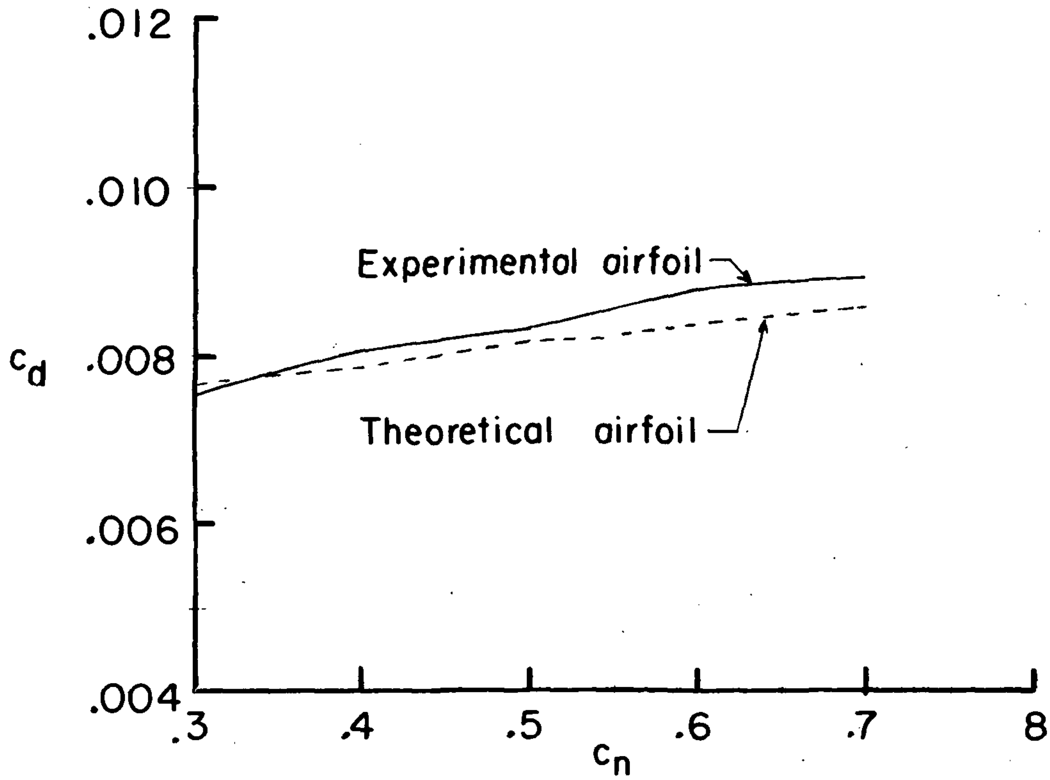
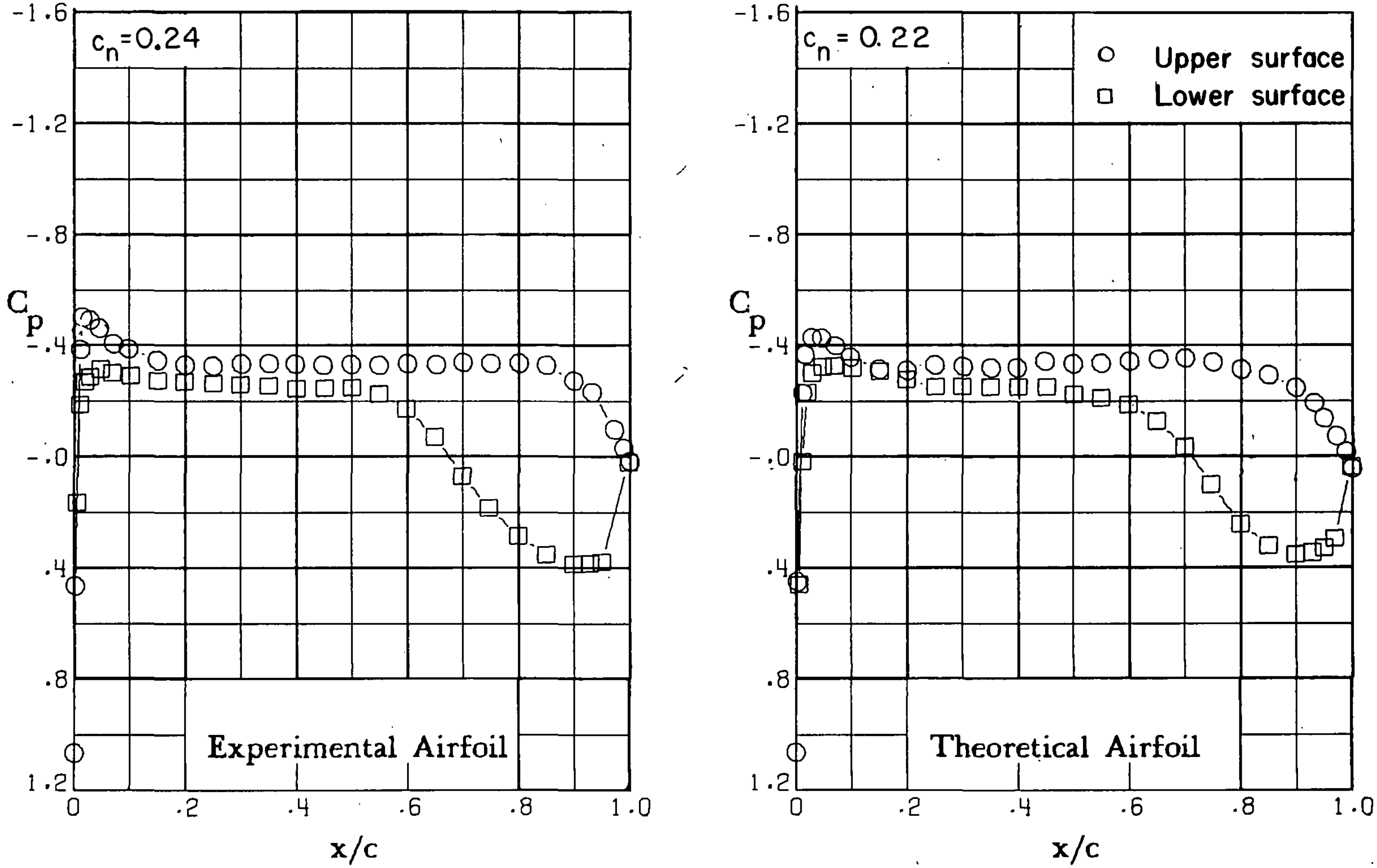


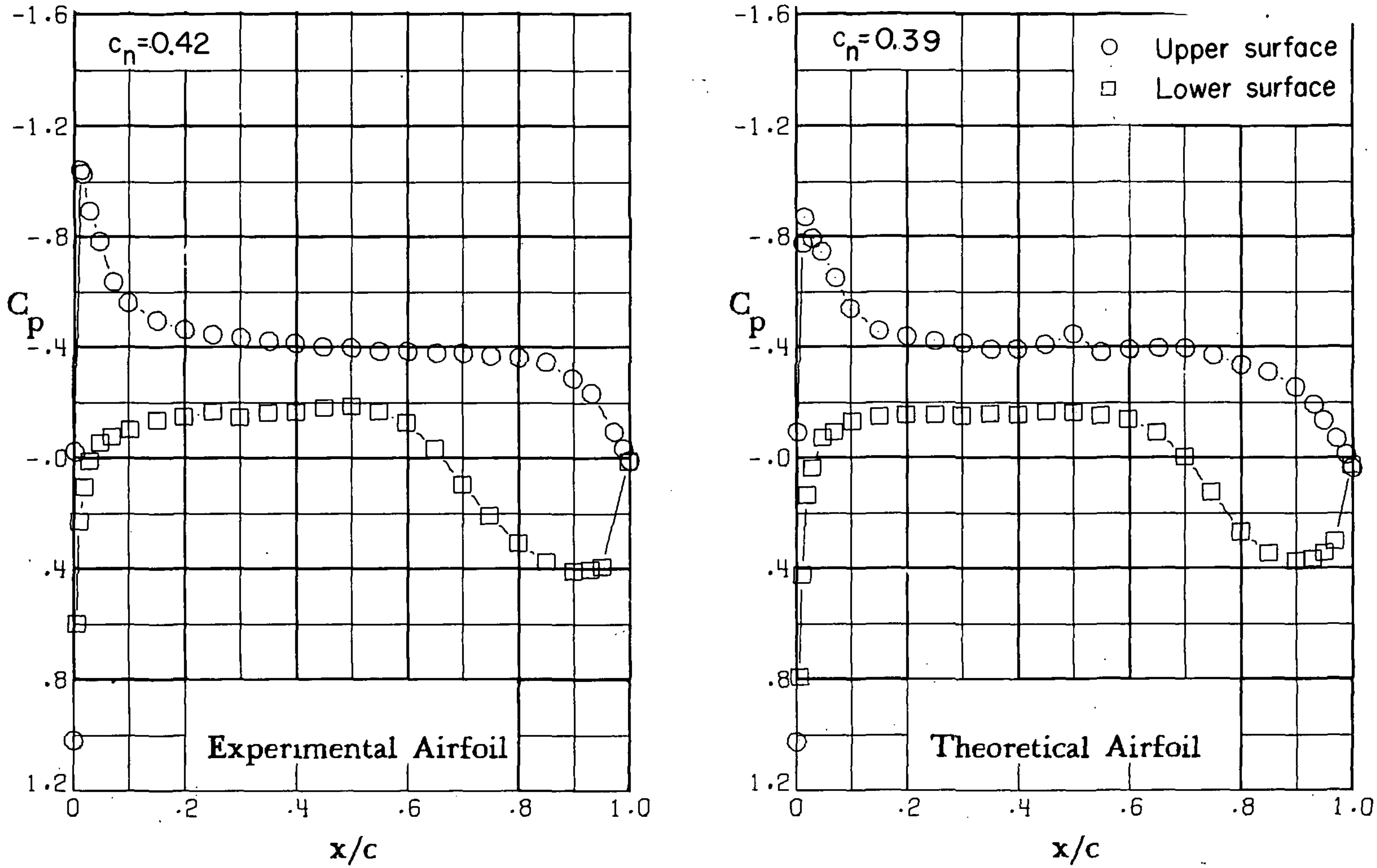
Figure 15.- Subcritical drag levels. M = 0.50.



(a) M = 0.50; α = -0.5°.

Figure 16.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

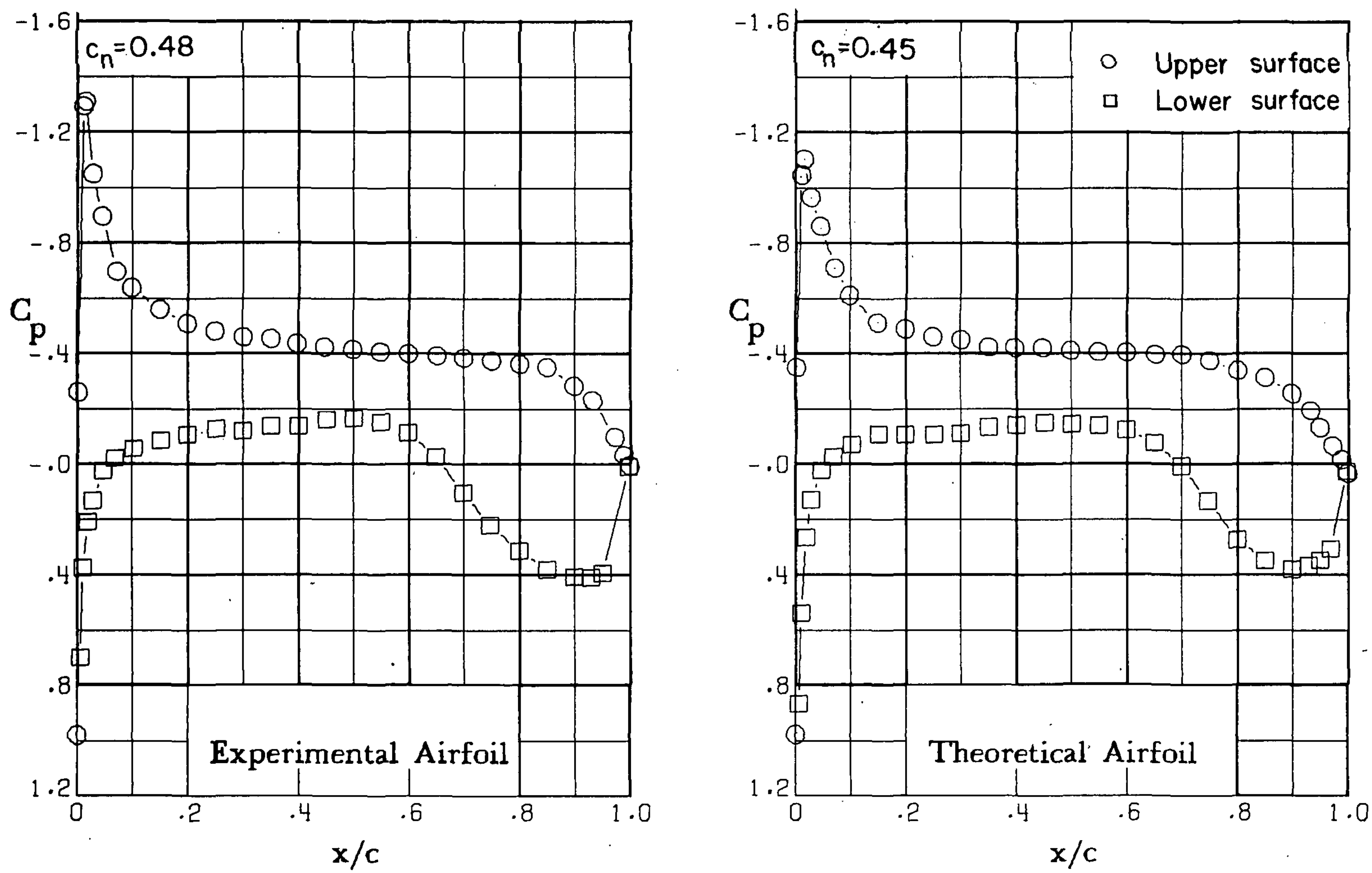
[redacted]



(b) M = 0.50; α = 1.0°.

Figure 16.- Continued.

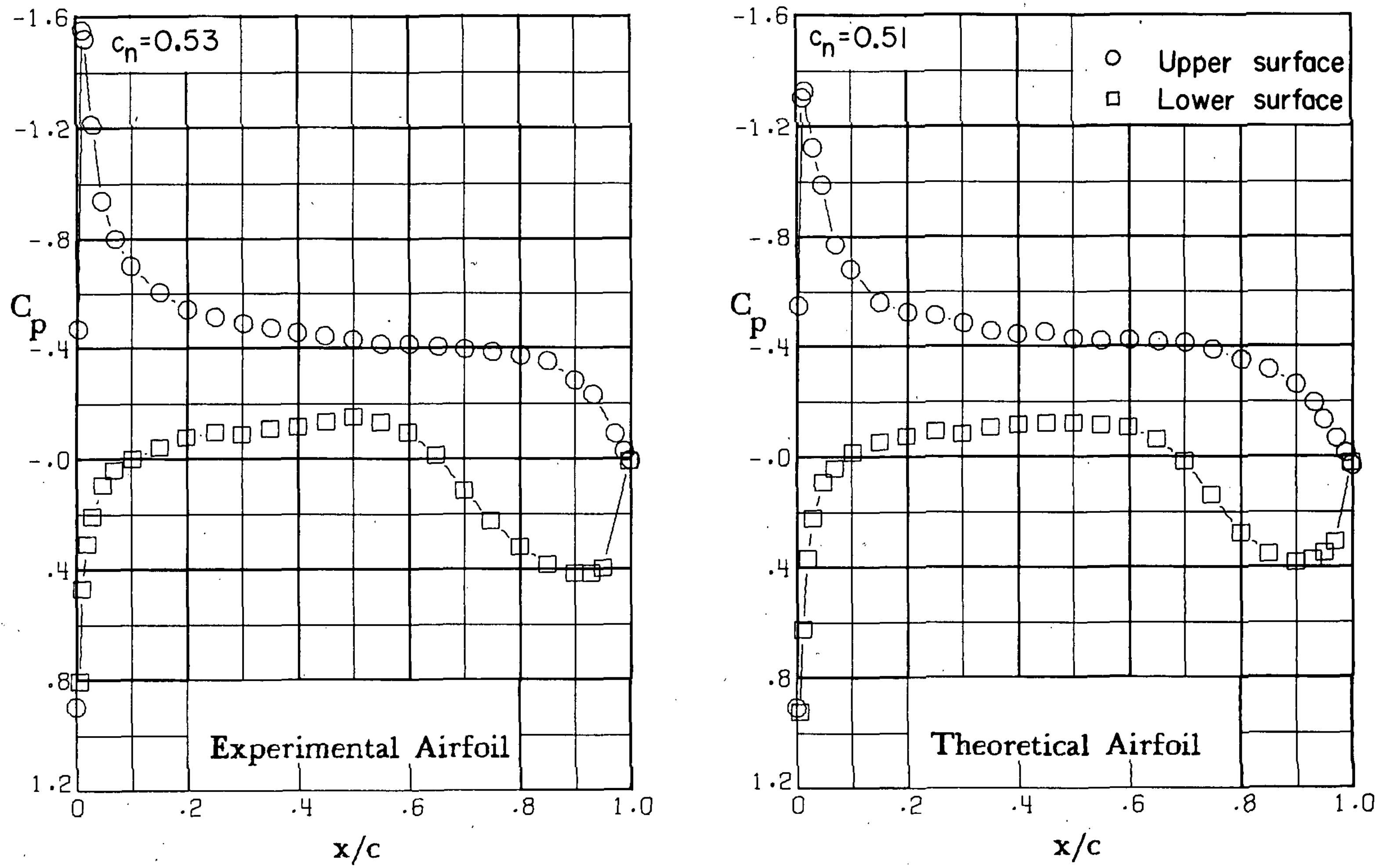
[redacted]



(c) $M = 0.50$; $\alpha = 1.5^{\circ}$.

Figure 16.- Continued.

[redacted]

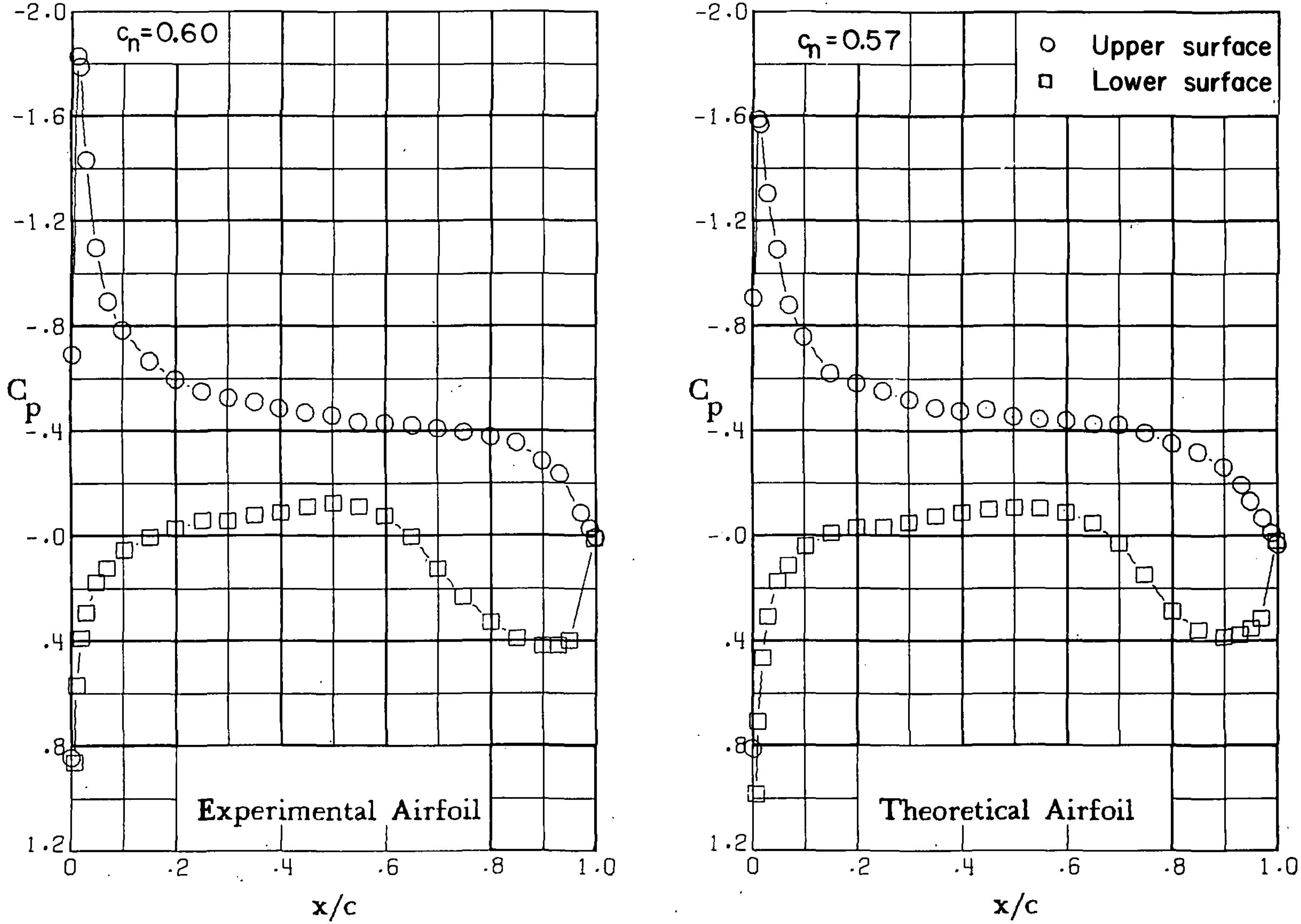


(d) M = 0.50; α = 2.0°.

Figure 16.- Continued.

[redacted]

[redacted]

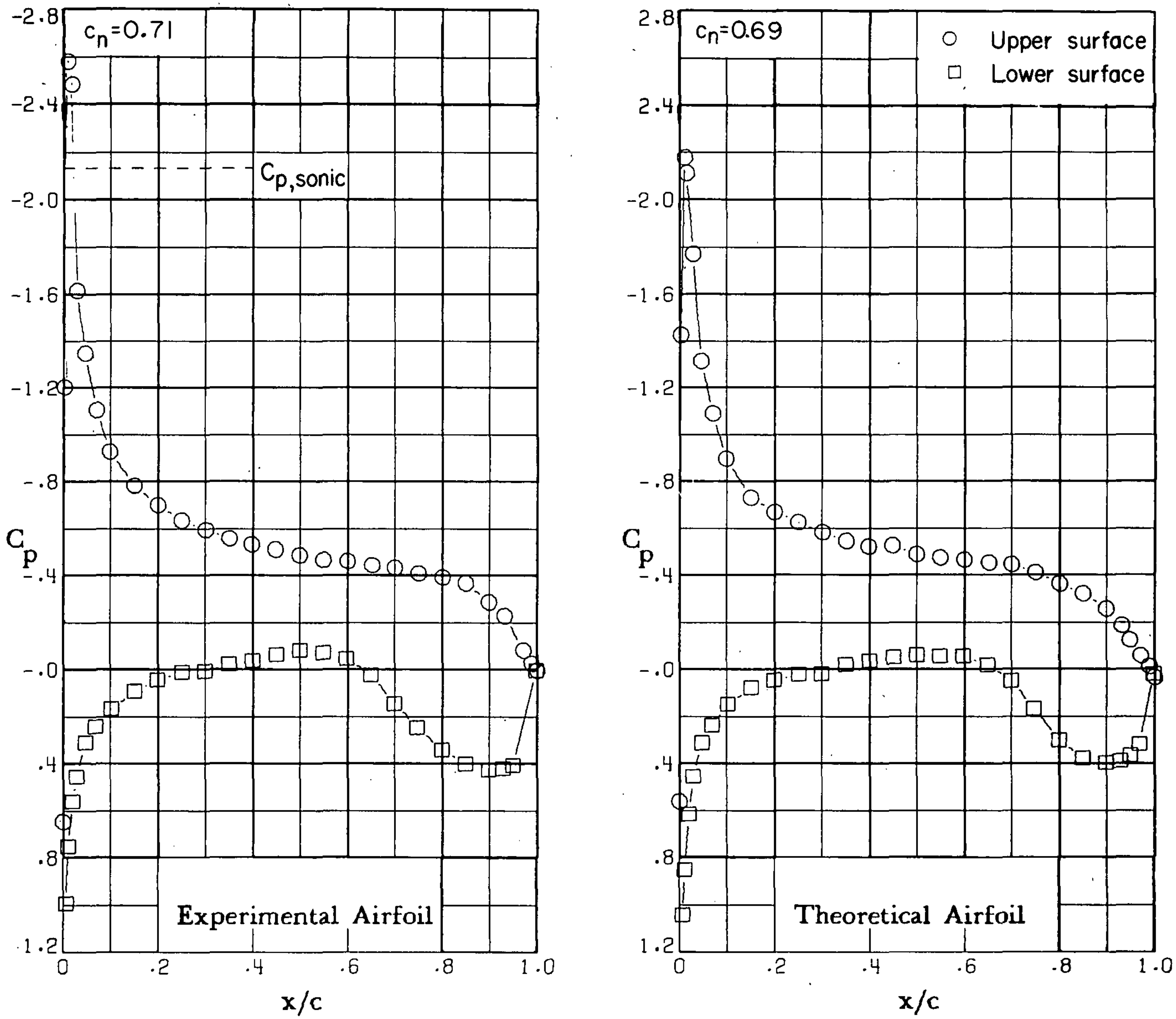


(e) M = 0.50; α = 2.5°.

Figure 16.- Continued.

[redacted]

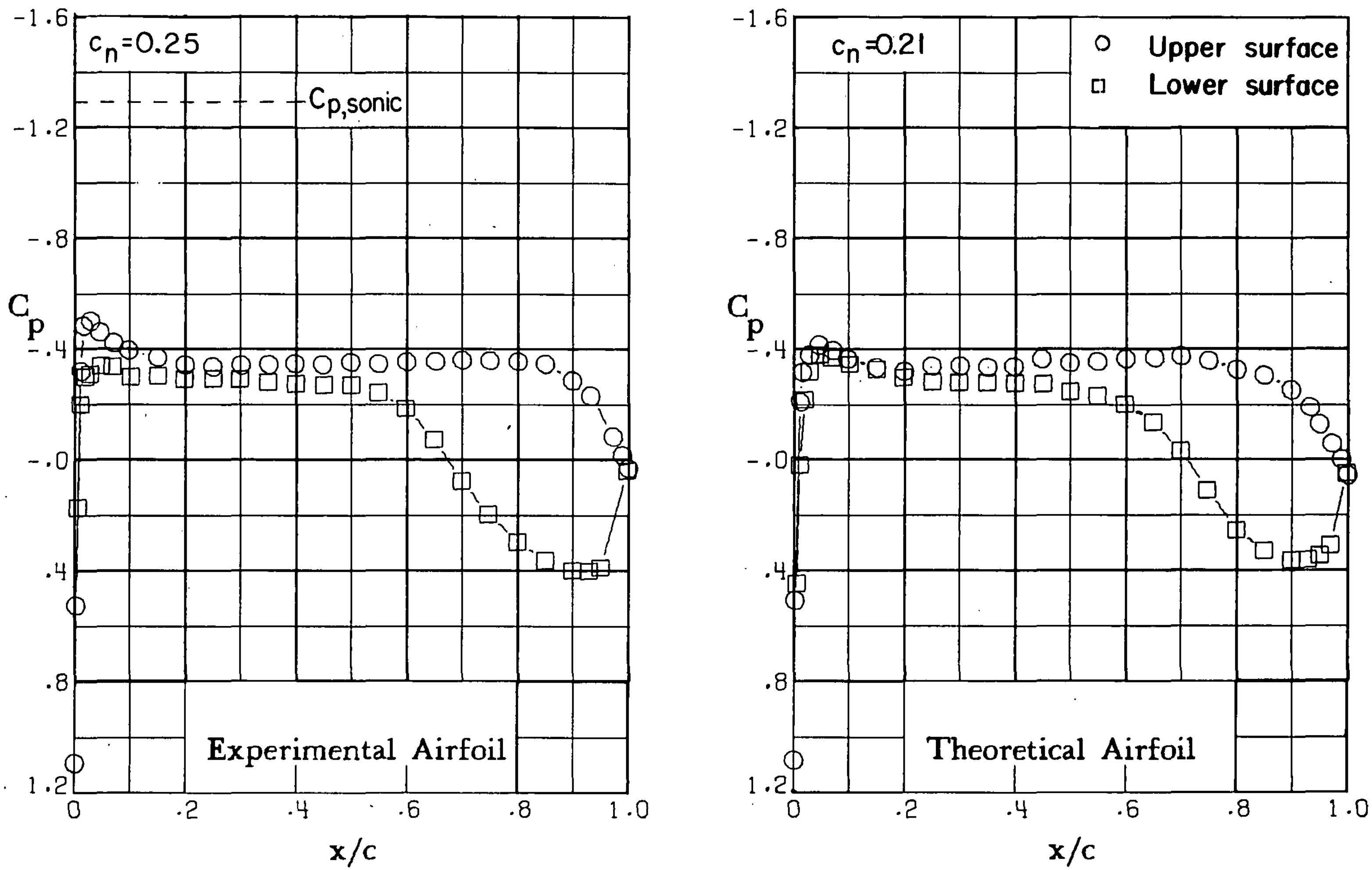
[redacted]



(f) M = 0.50; α = 3.5°.

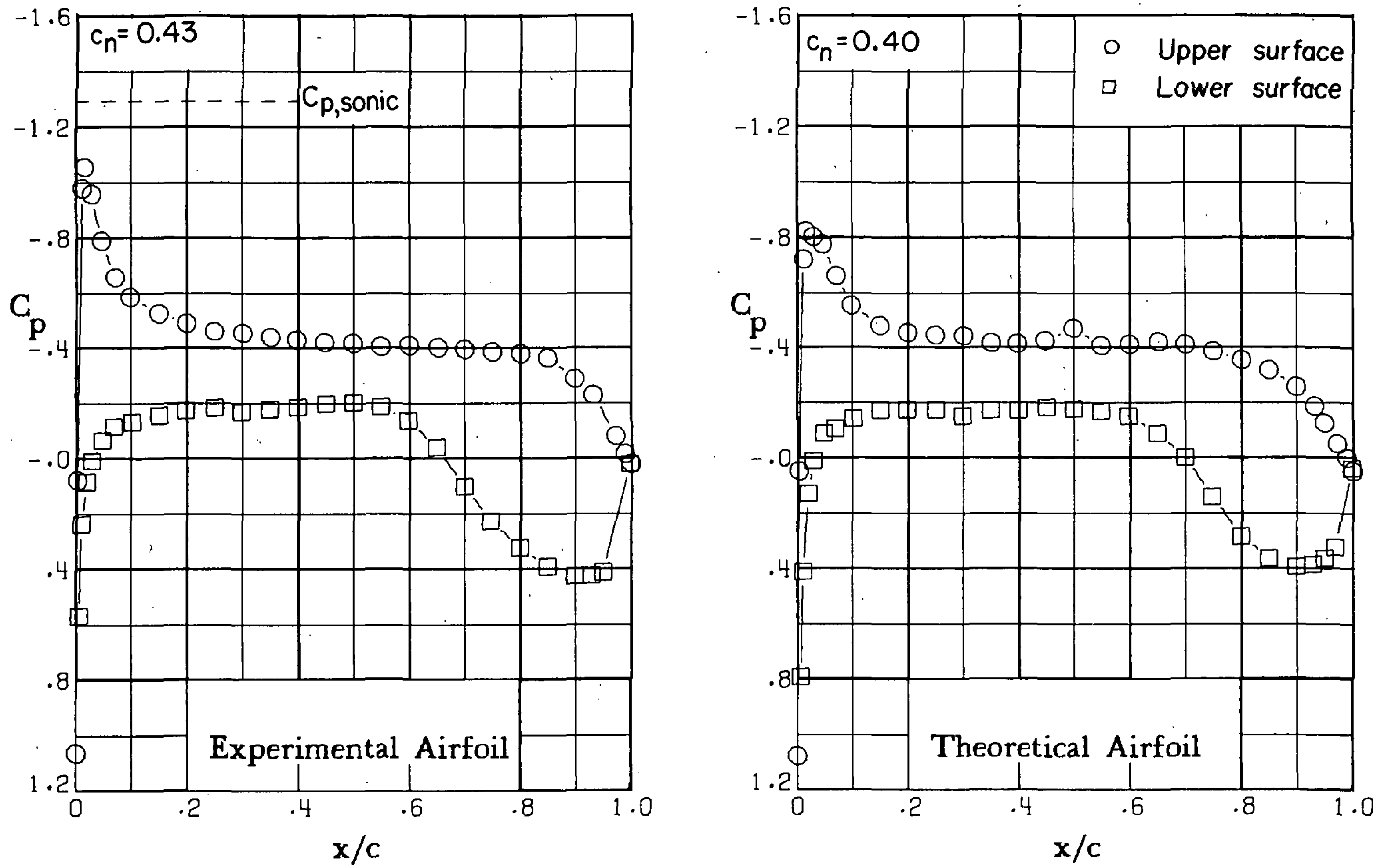
Figure 16.- Concluded.

[redacted]



(a) M = 0.60; α = -0.5°.

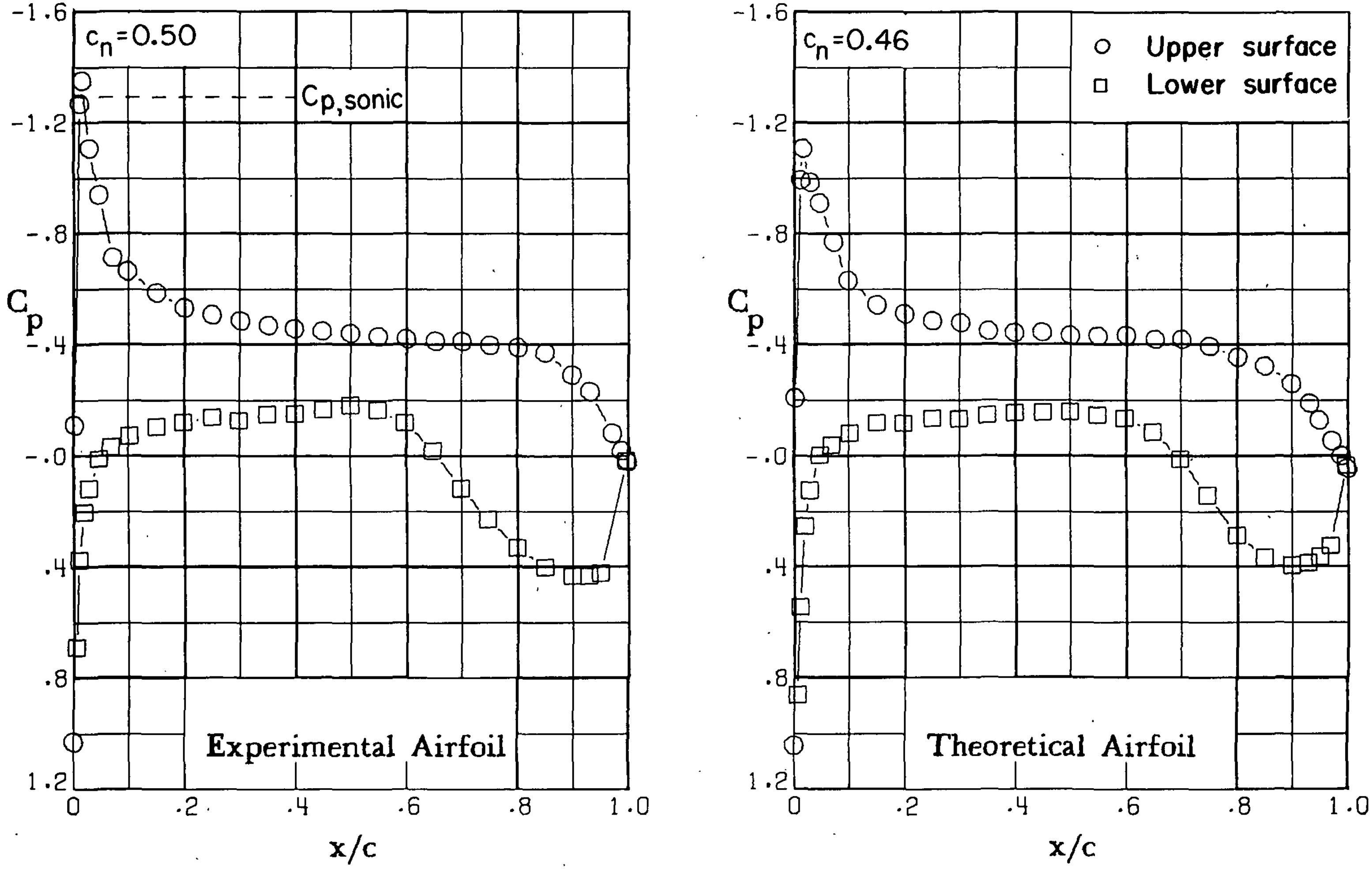
Figure 17.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.



(b) M = 0.60; α = 1.0°.

Figure 17.- Continued.

[redacted]

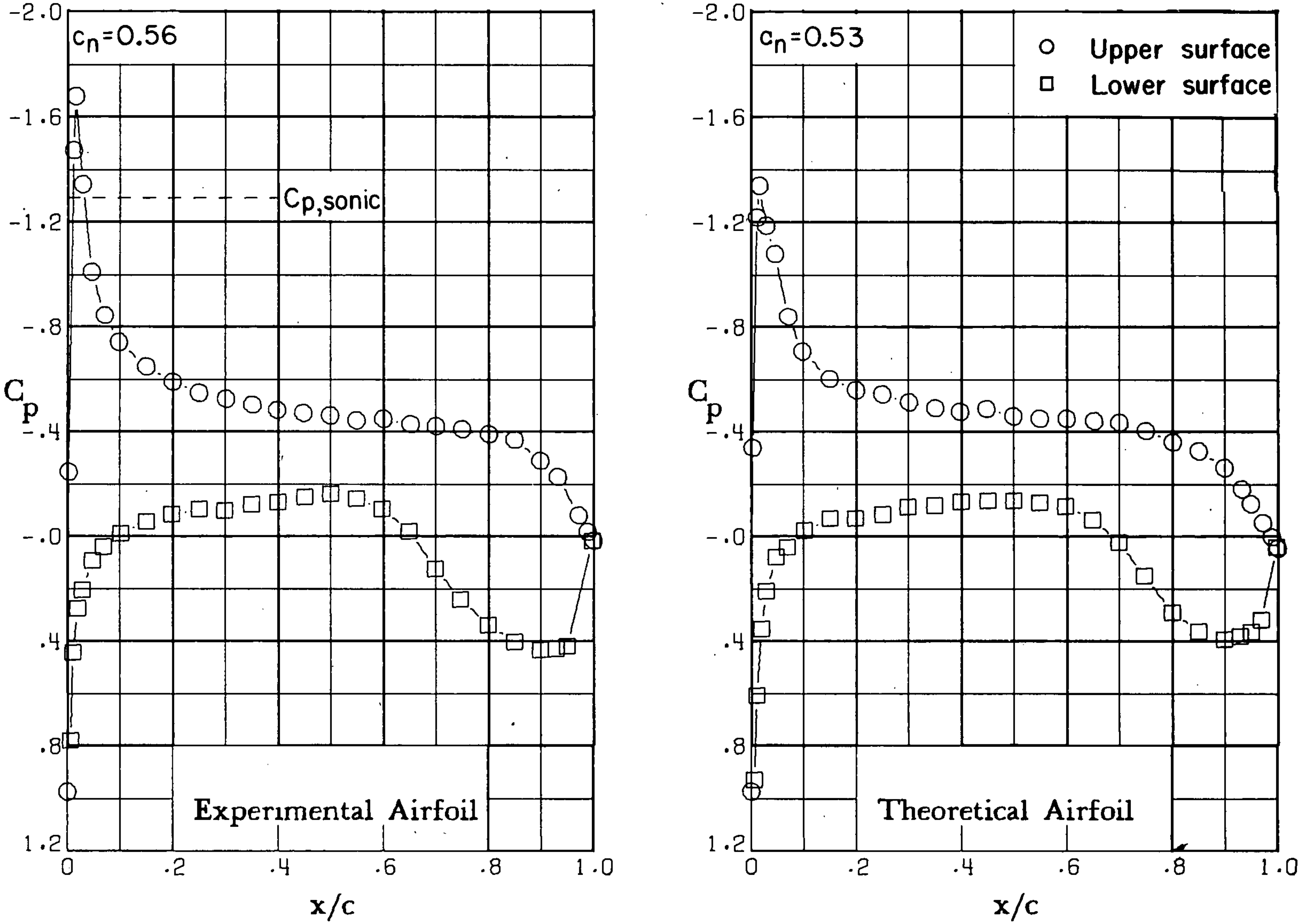


(c) M = 0.60; α = 1.5°.

Figure 17.- Continued.

[redacted]

[redacted]

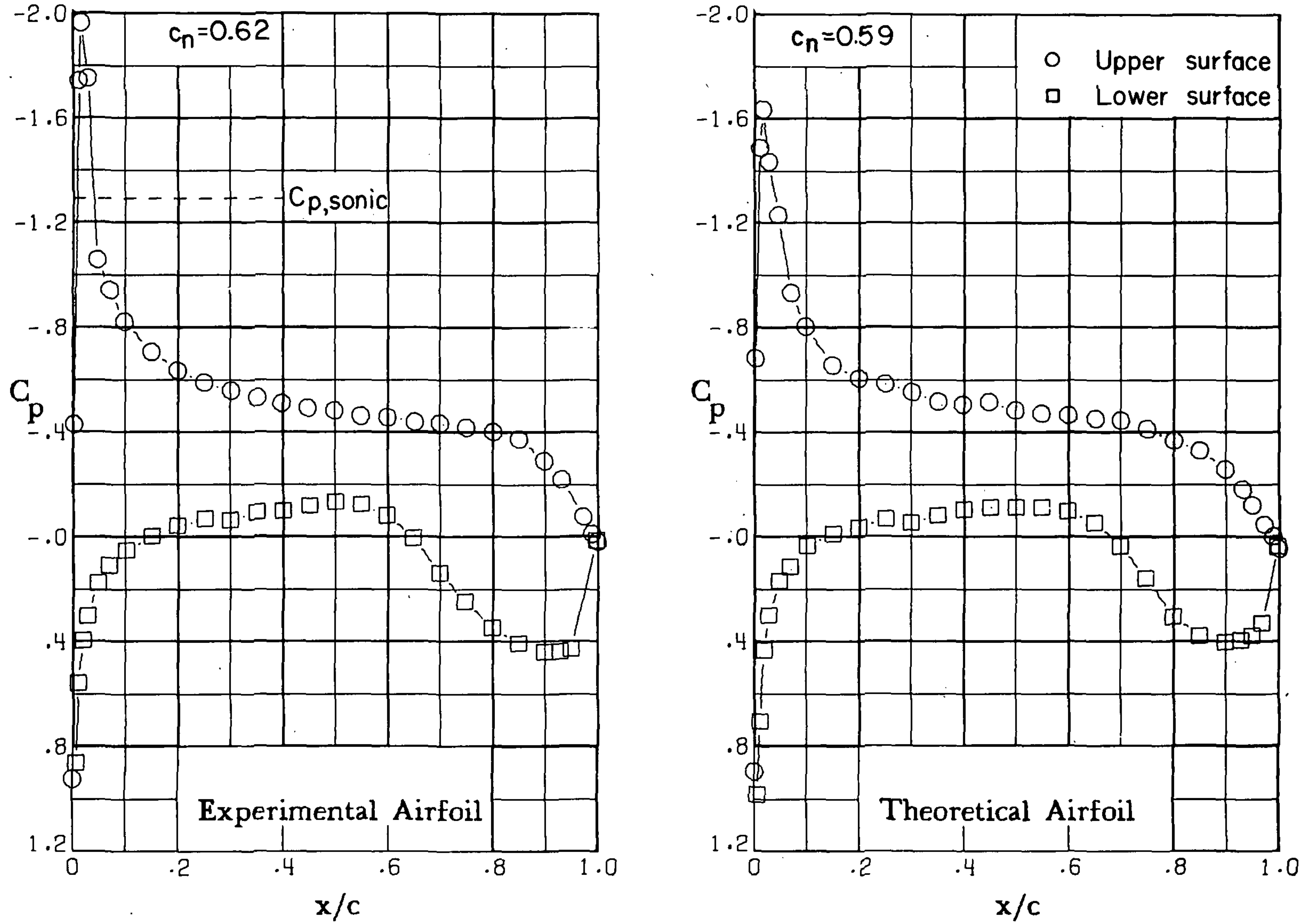


(d) M = 0.60; $\alpha = 2.0^{o}$.

Figure 17.- Continued.

[redacted]

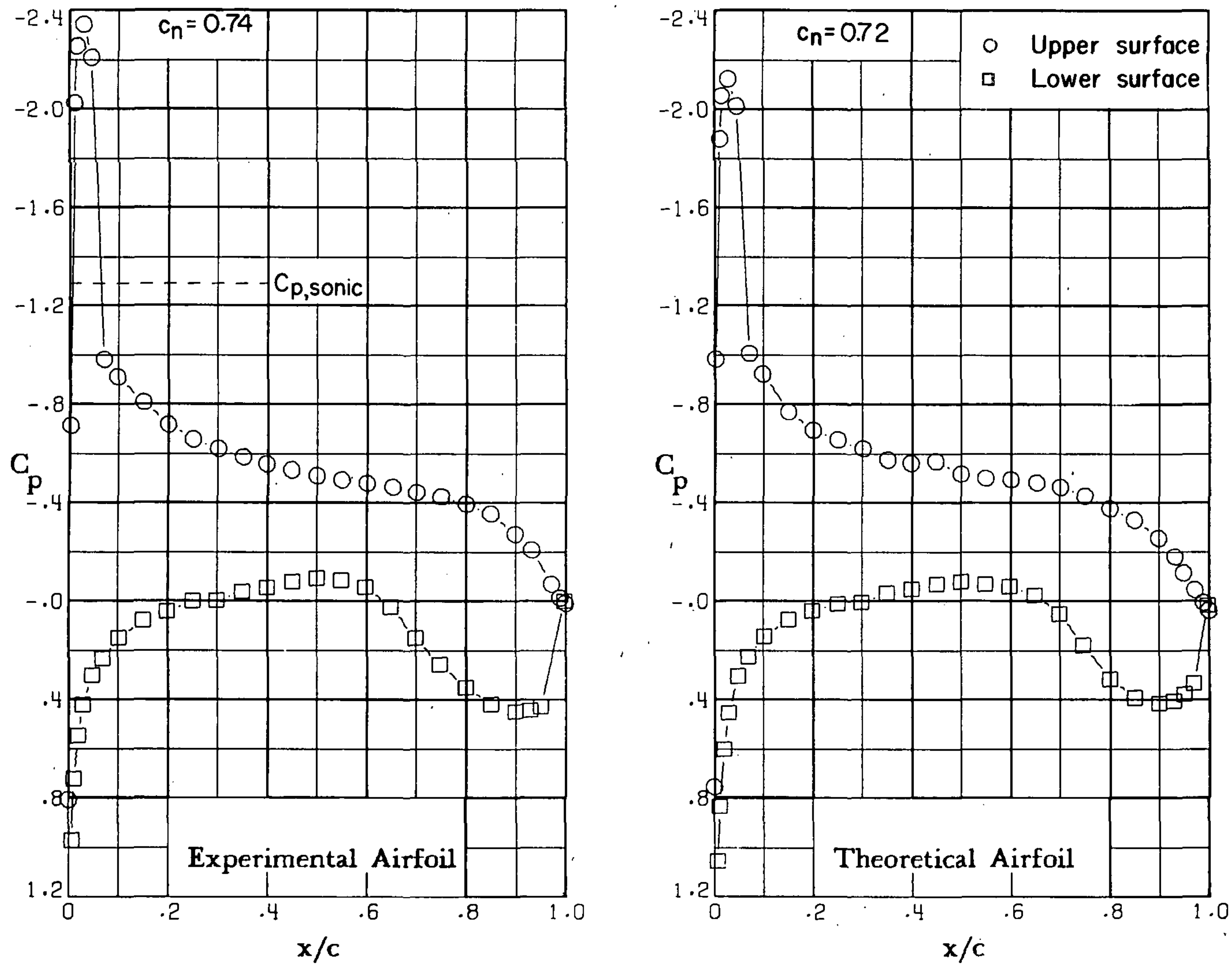
[redacted]



(e) $M = 0.60$; $\alpha = 2.5^o$.

Figure 17.- Continued.

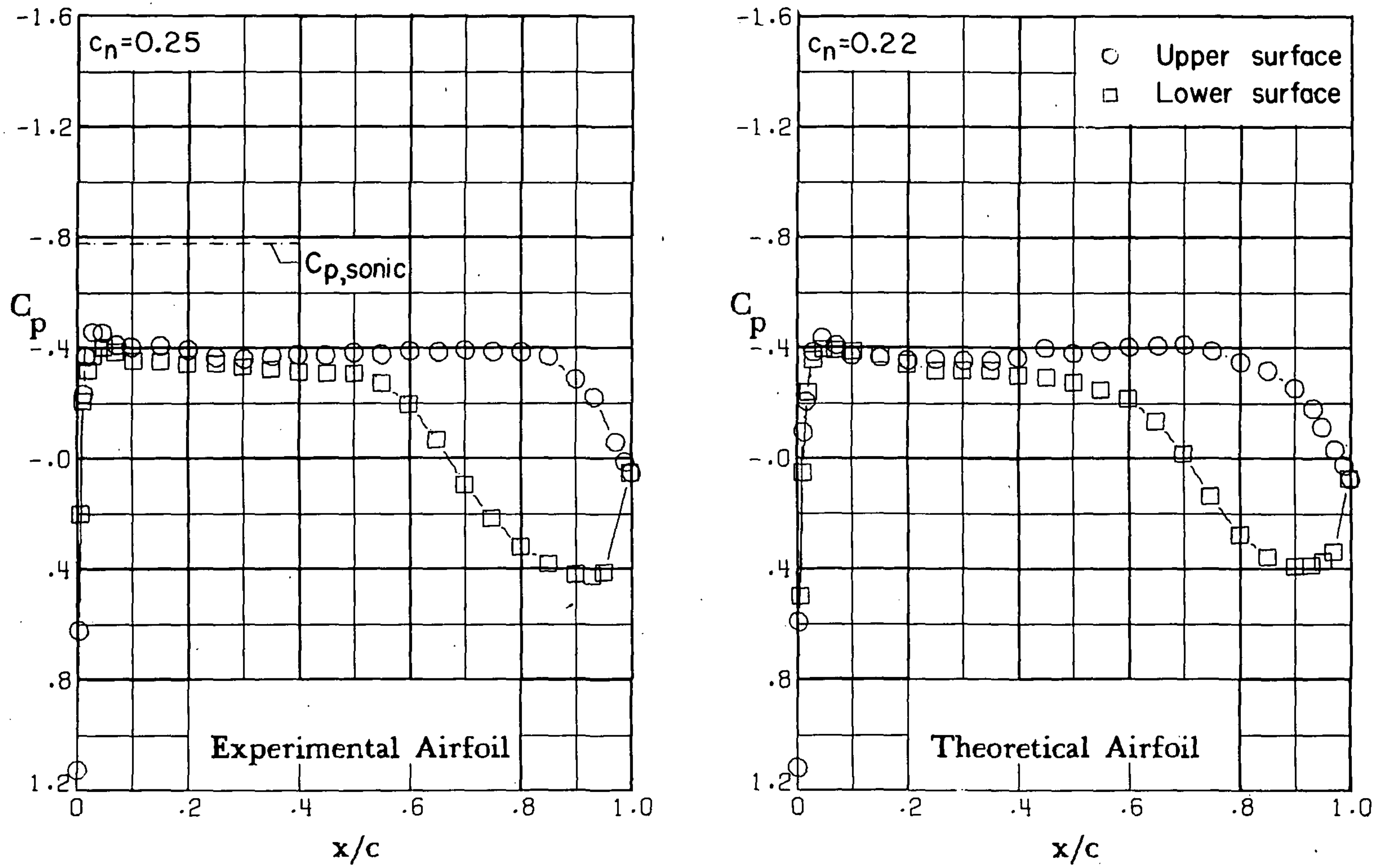
[redacted]



(f) M = 0.60; α = 3.5°.

Figure 17.- Concluded.

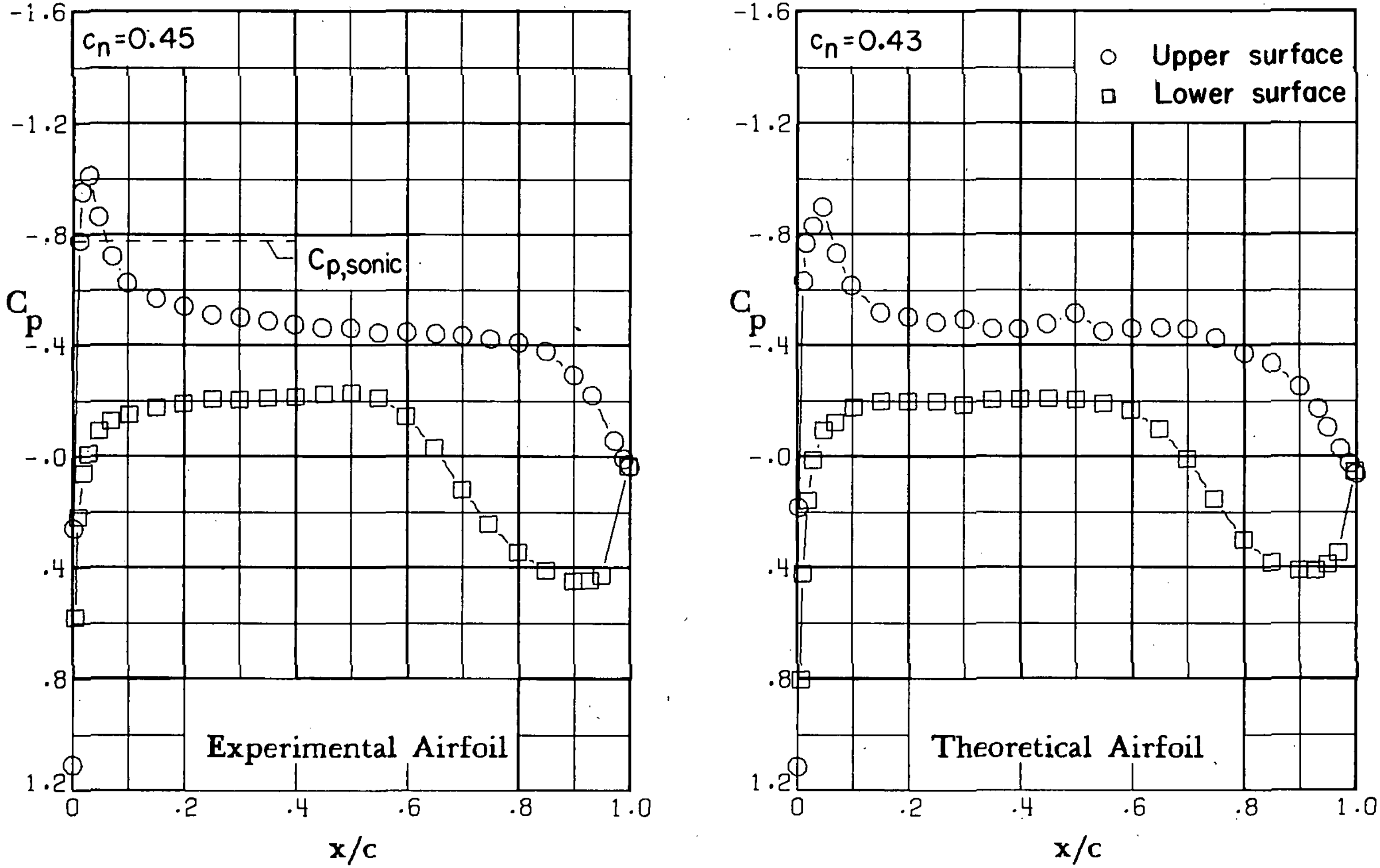
[redacted]



(a) M = 0.70; α = -0.5°.

Figure 18.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

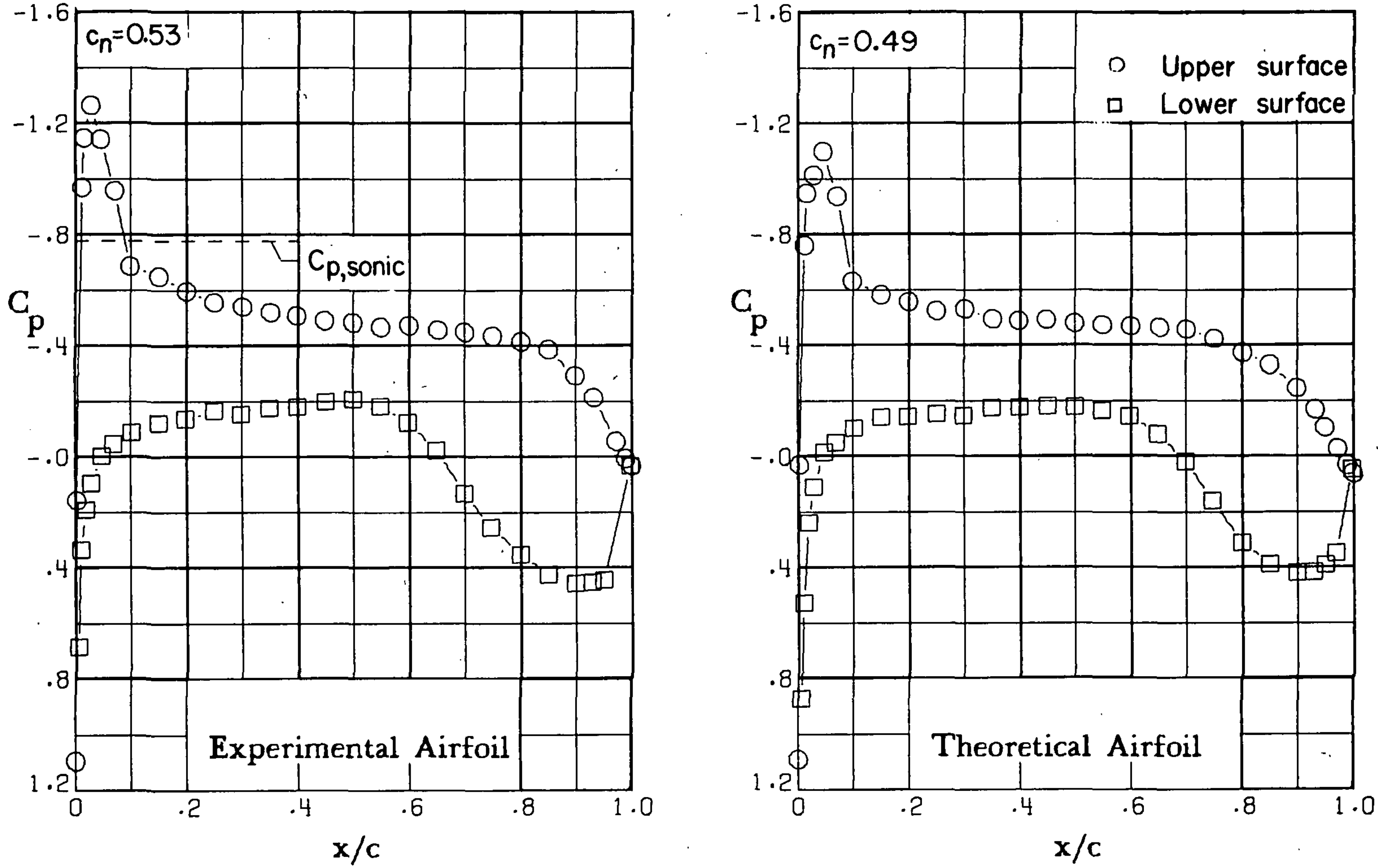
[redacted]



(b) $M = 0.70$; $\alpha = 1.0^o$.

Figure 18.- Continued.

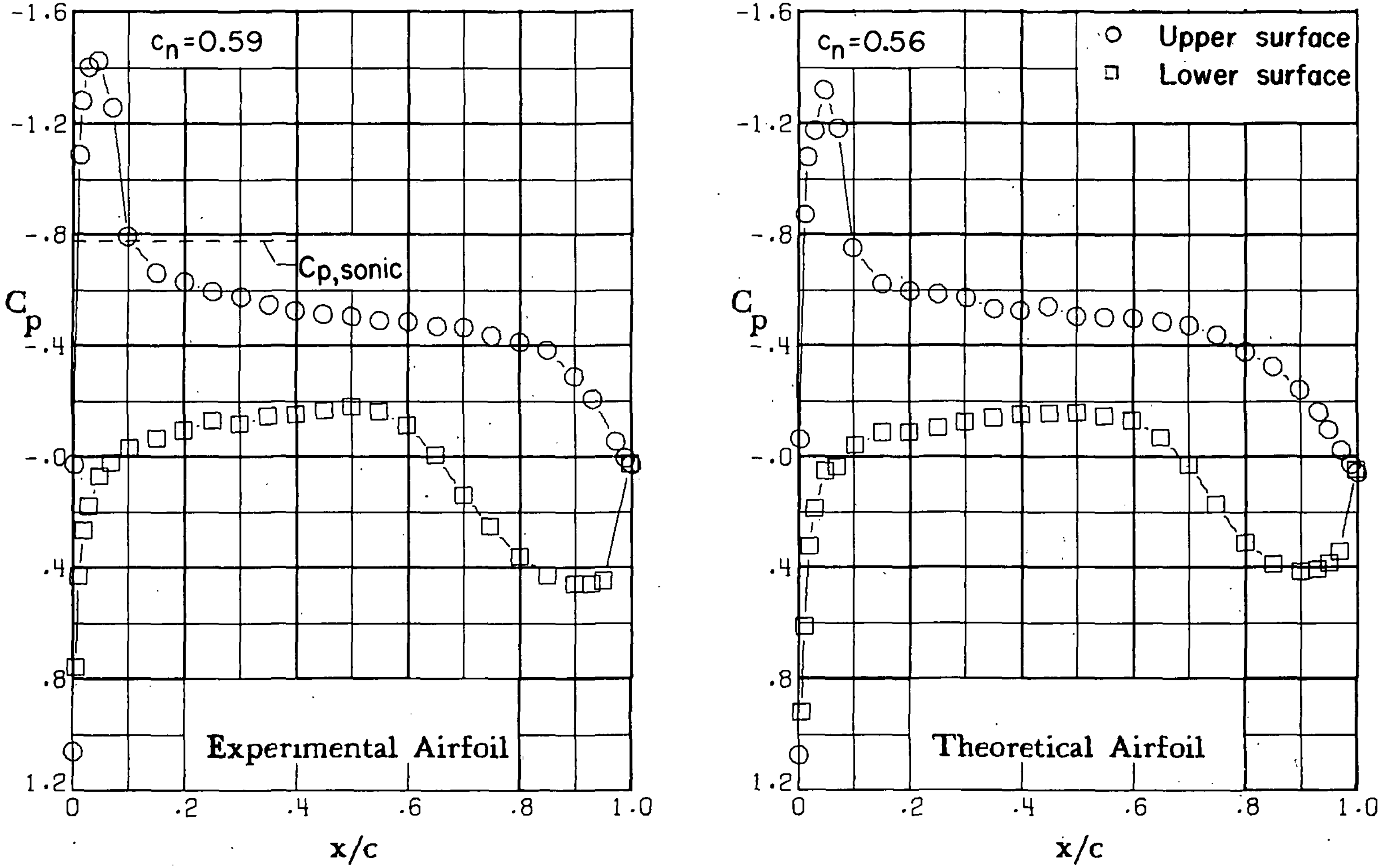
[redacted]



(c) $M = 0.70$; $\alpha = 1.5^{\circ}$.

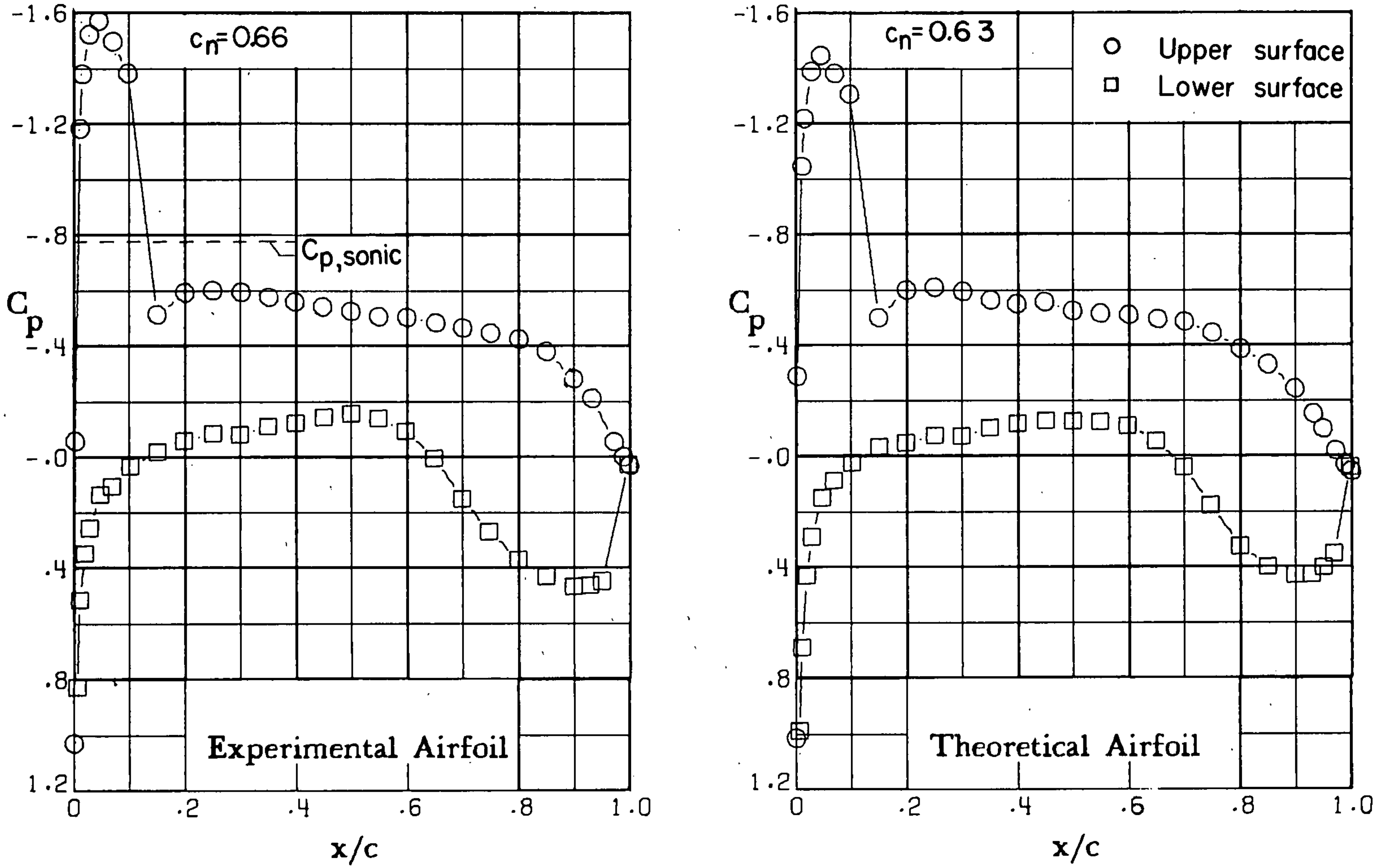
Figure 18.- Continued.

[redacted]



(d) M = 0.70; α = 2.0°.

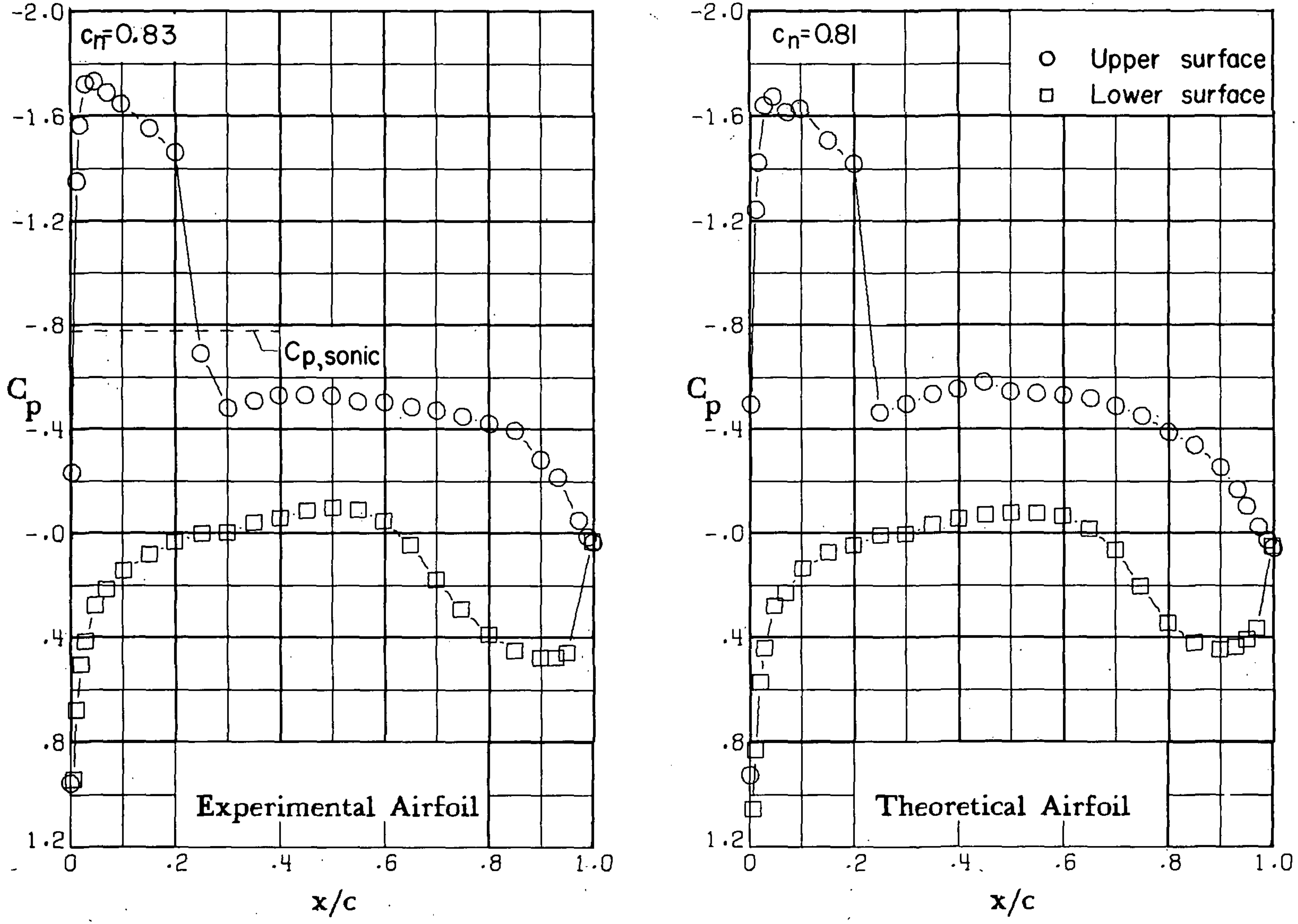
Figure 18.- Continued.



(e) M = 0.70; $\alpha = 2.5^{\circ}$.

Figure 18.- Continued.

[redacted]

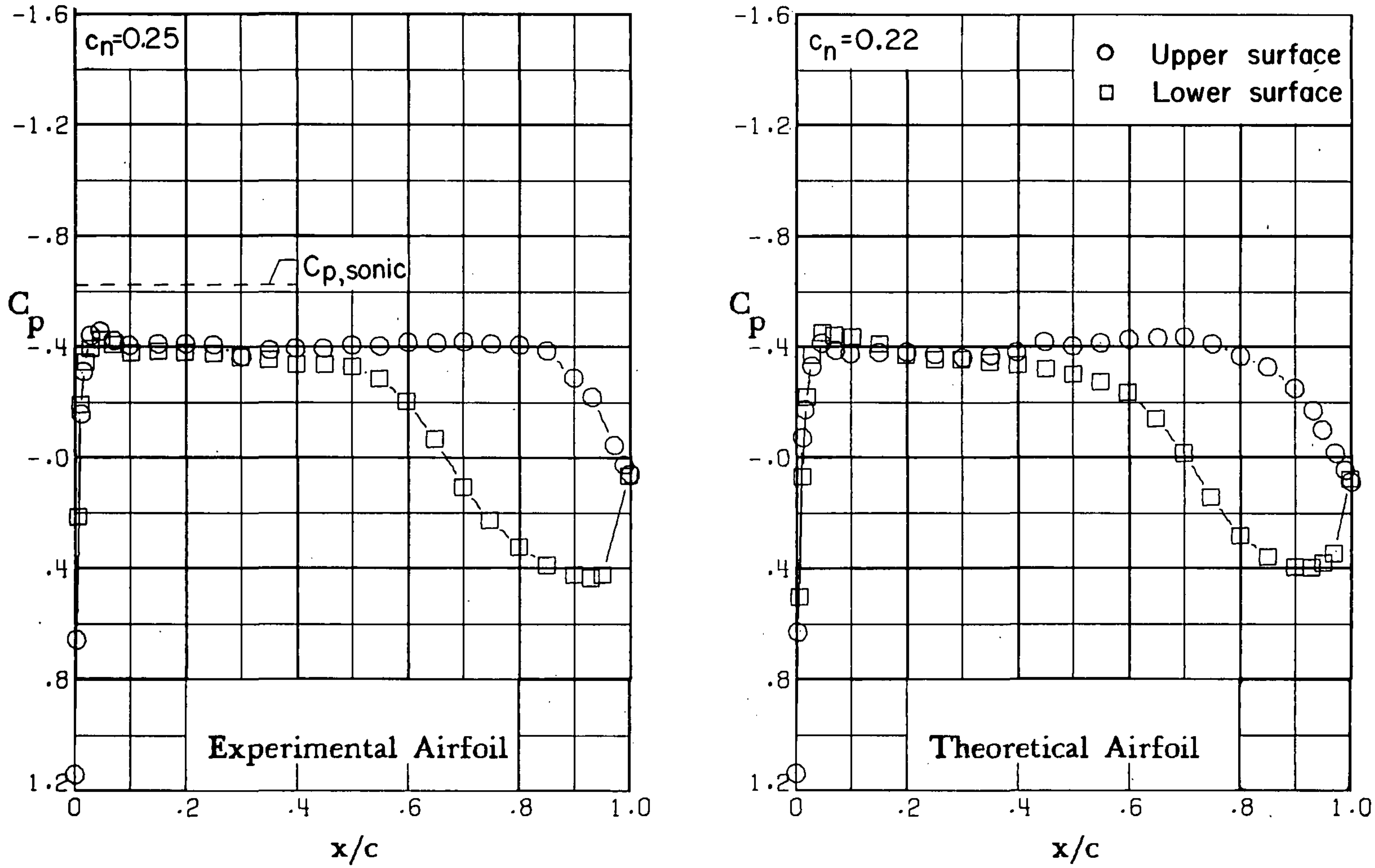


(f) M = 0.70; $\alpha = 3.5^{o}$.

Figure 18.- Concluded.

[redacted]

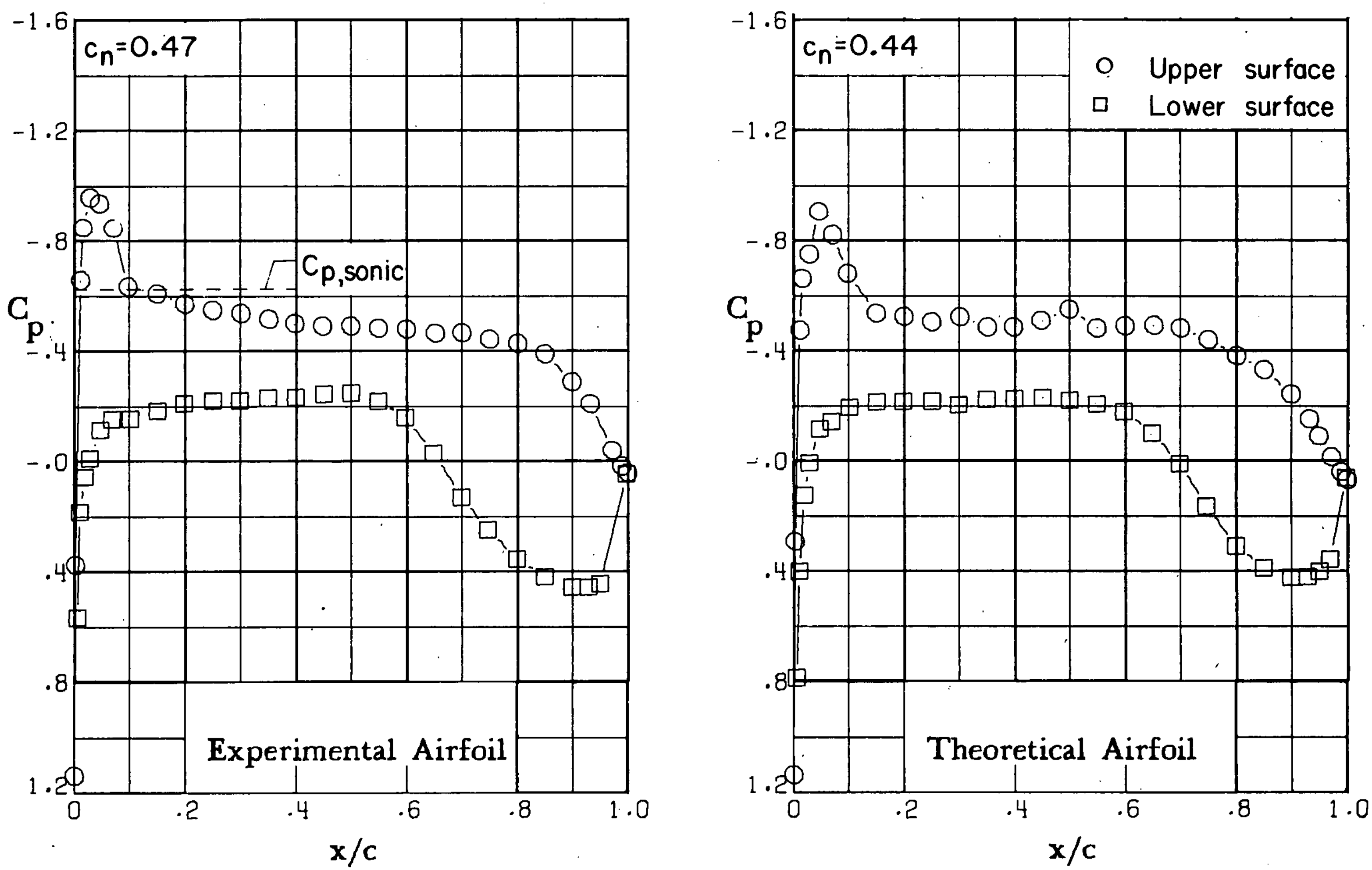
[redacted]



(a) M = 0.74; α = -0.5°.

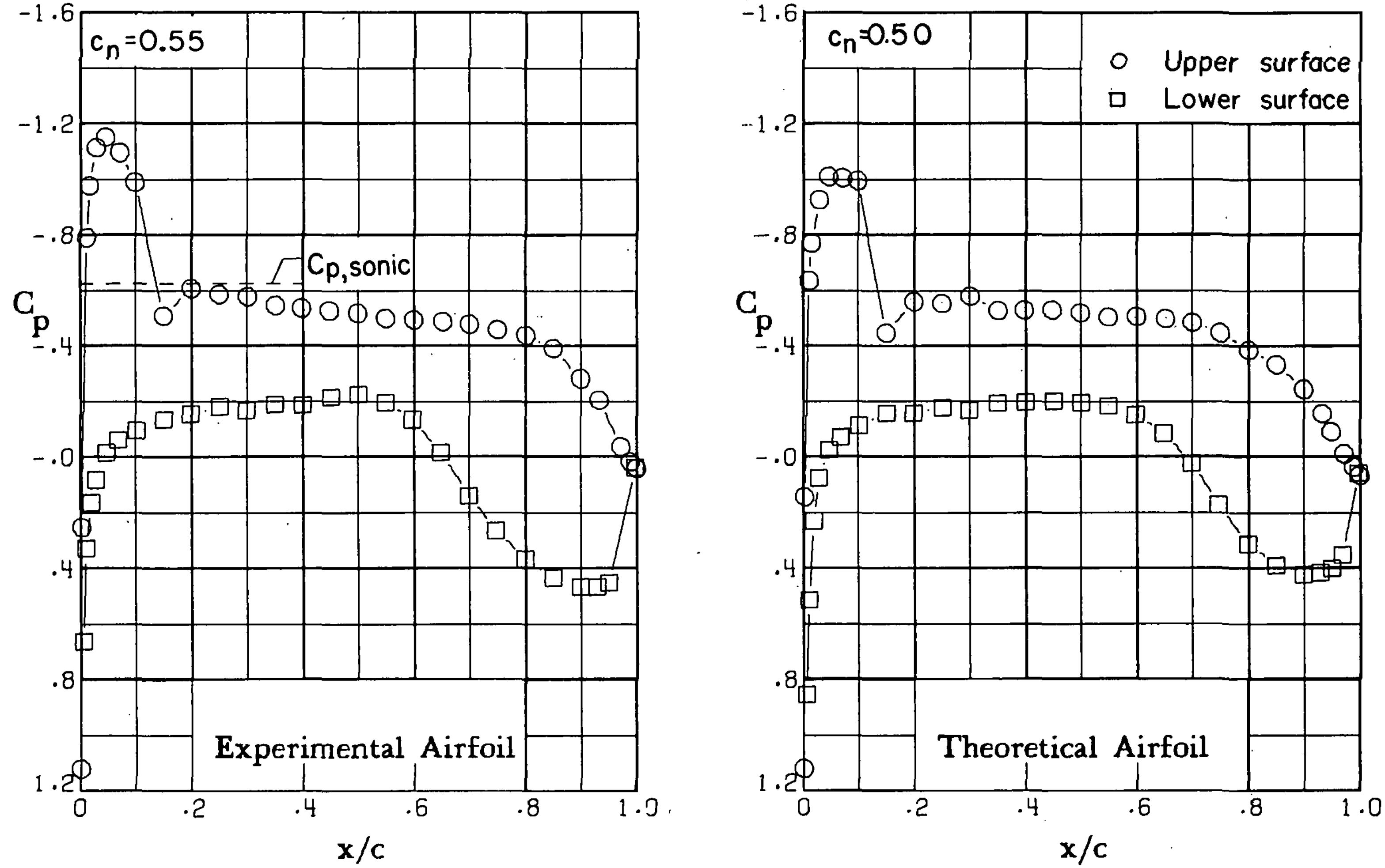
Figure 19.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

[redacted]



(b) $M = 0.74; \quad \alpha = 1.0^o$.

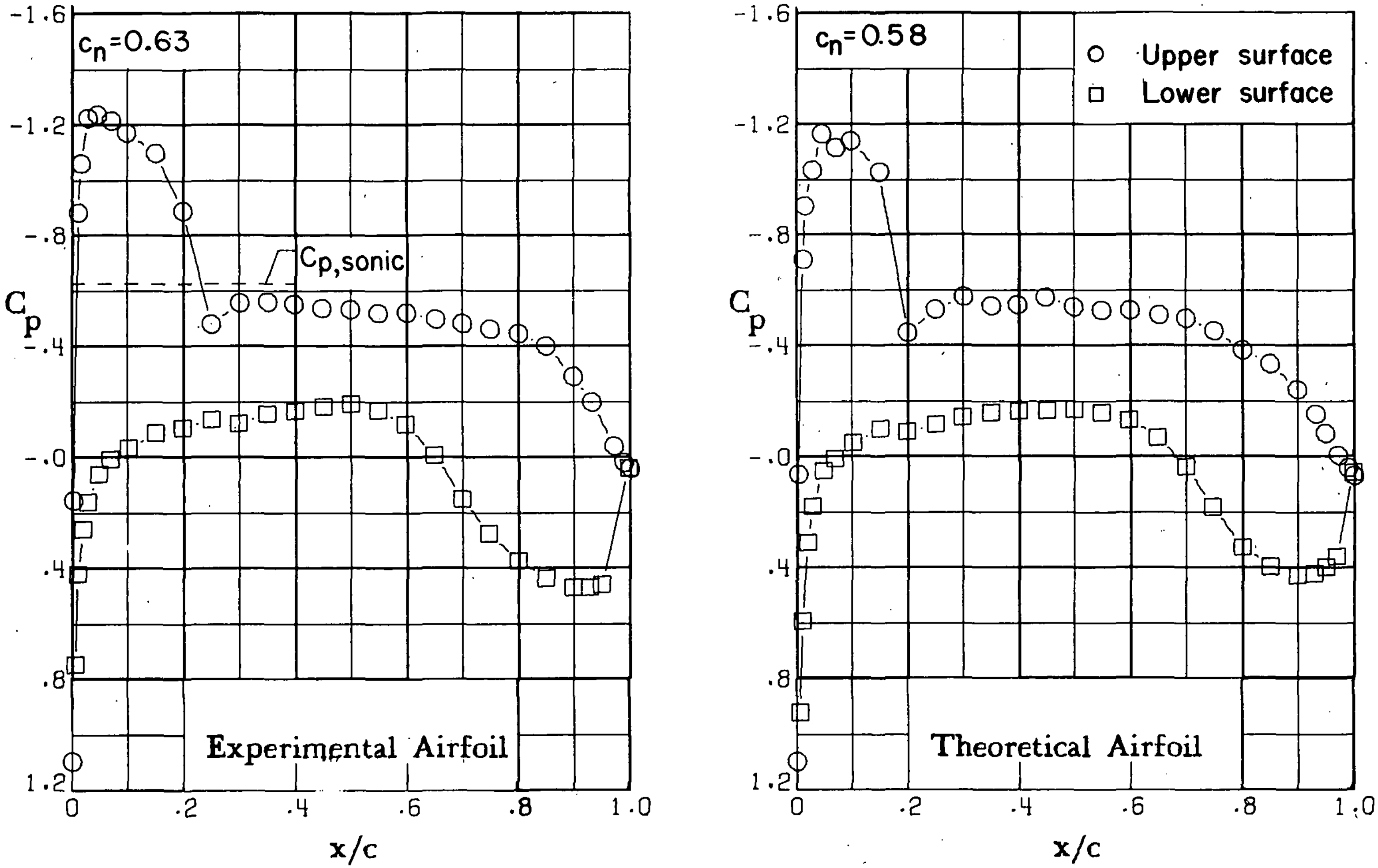
Figure 19.- Continued.



(c) M = 0.74; $\alpha = 1.5^{\circ}$.

Figure 19.- Continued.

[redacted]

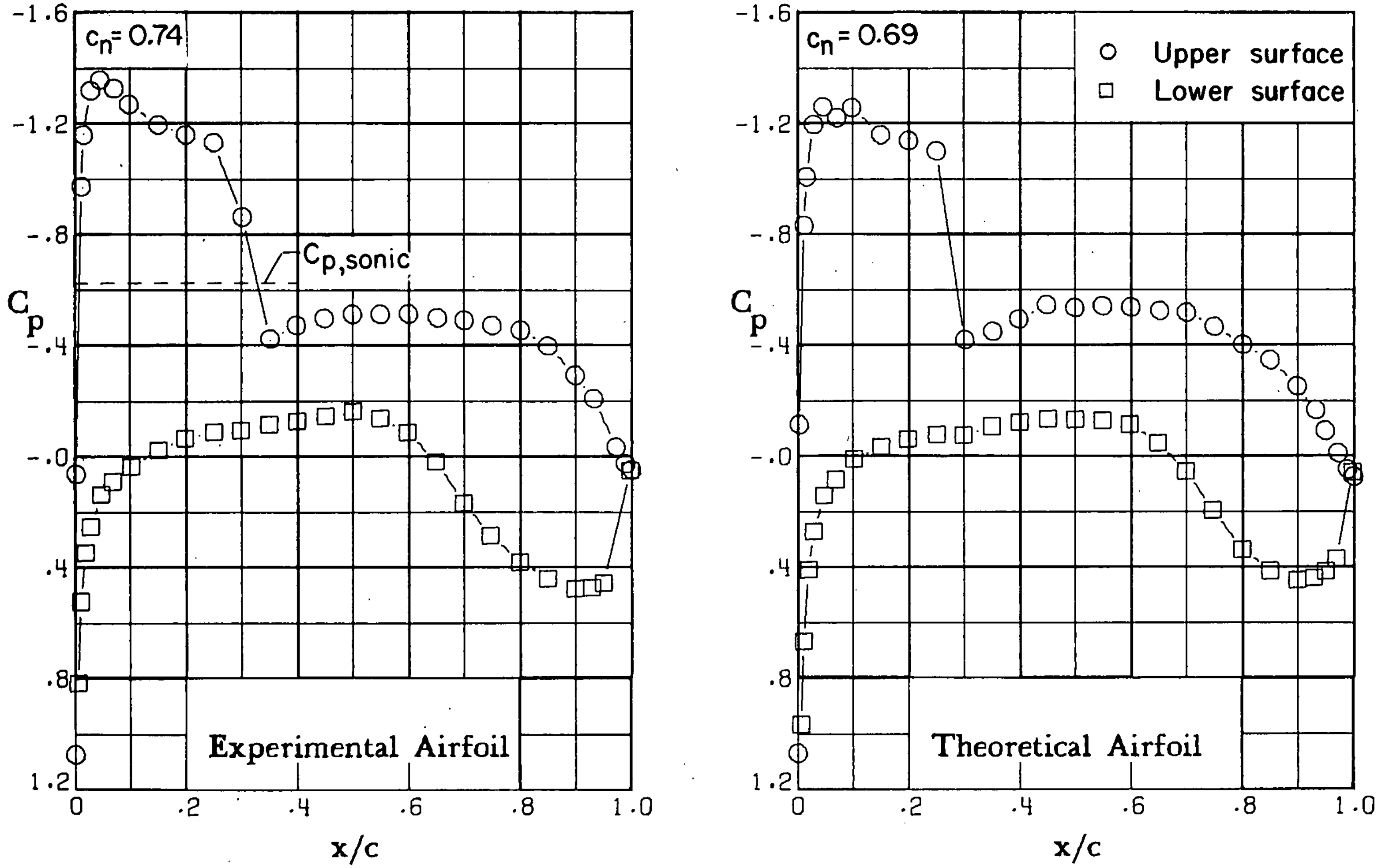


(d) M = 0.74; α = 2.0°.

Figure 19.- Continued.

[redacted]

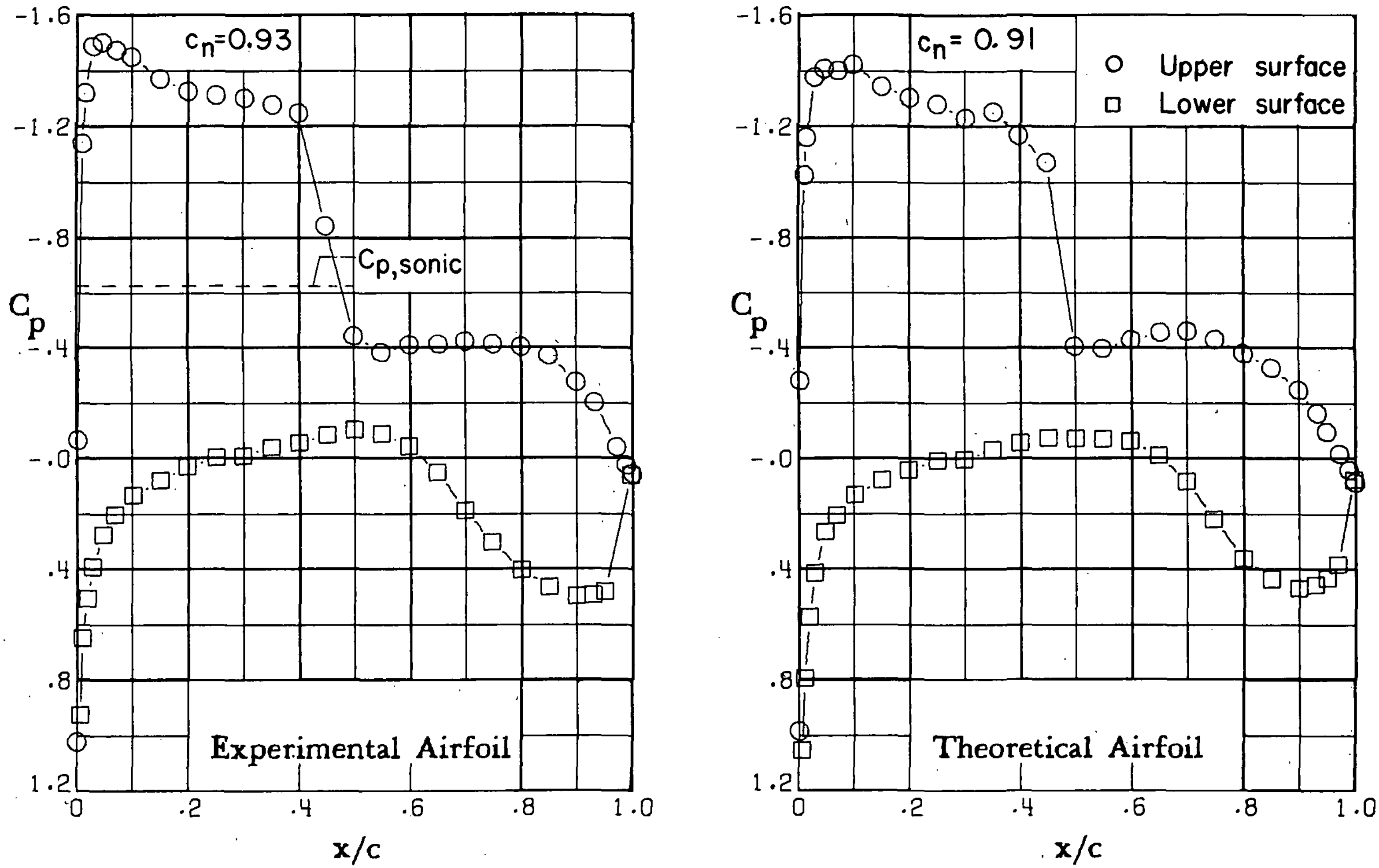
[redacted]



(e) M = 0.74; α = 2.5°.

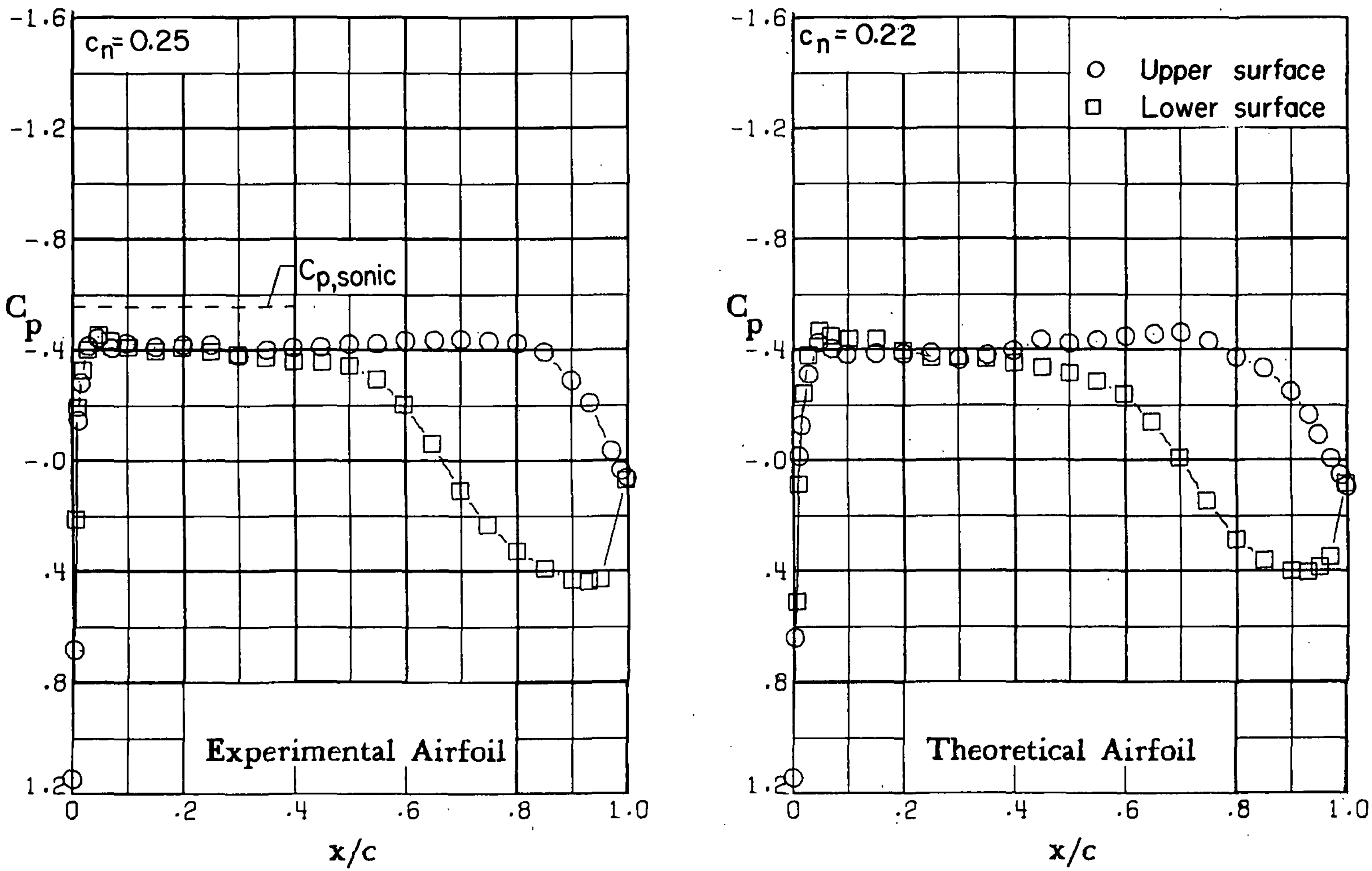
Figure 19.- Continued.

[redacted]



(f) $M = 0.74$; $\alpha = 3.5^{o}$.

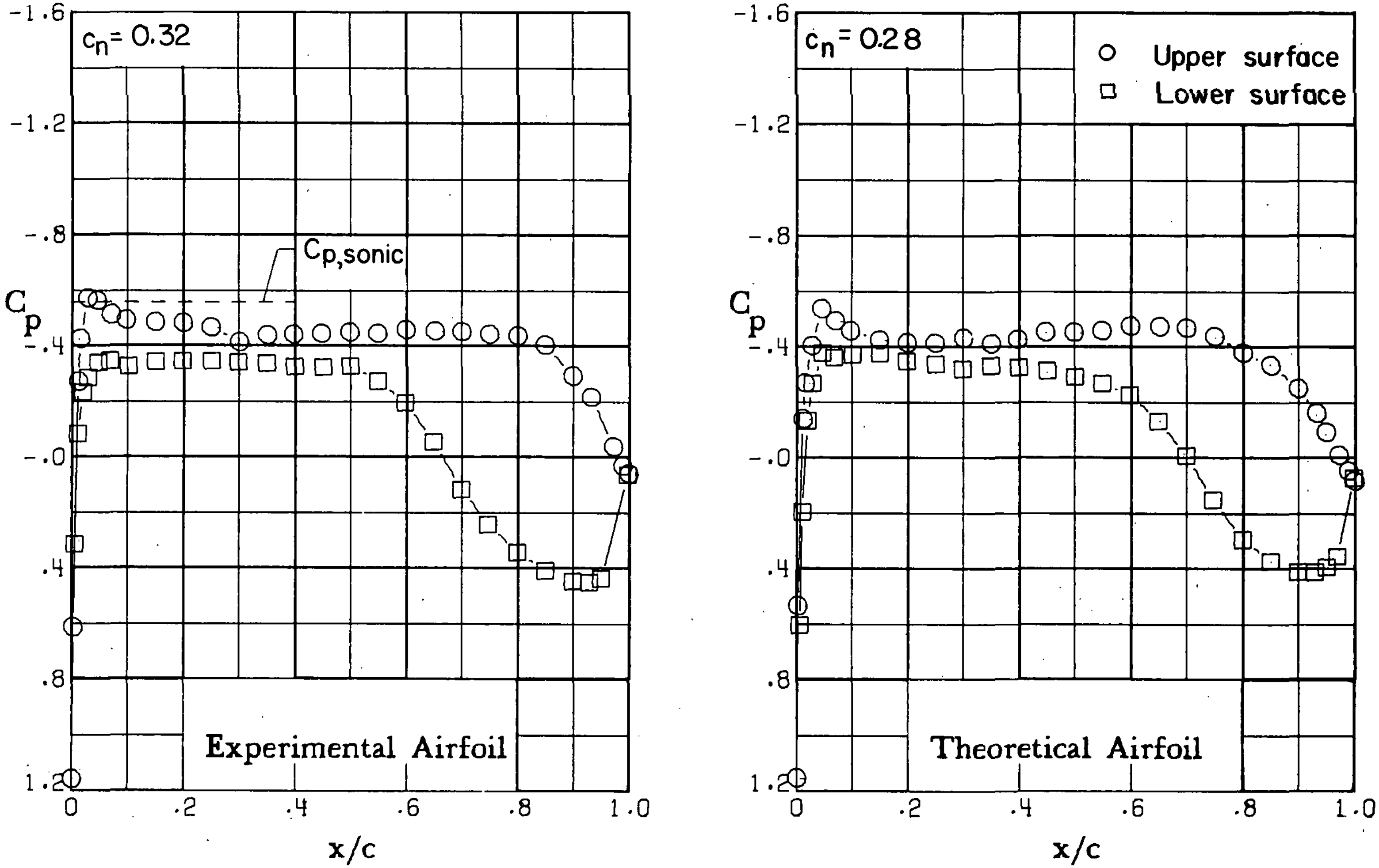
Figure 19.- Concluded.



(a) M = 0.76; α = -0.5°.

Figure 20.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

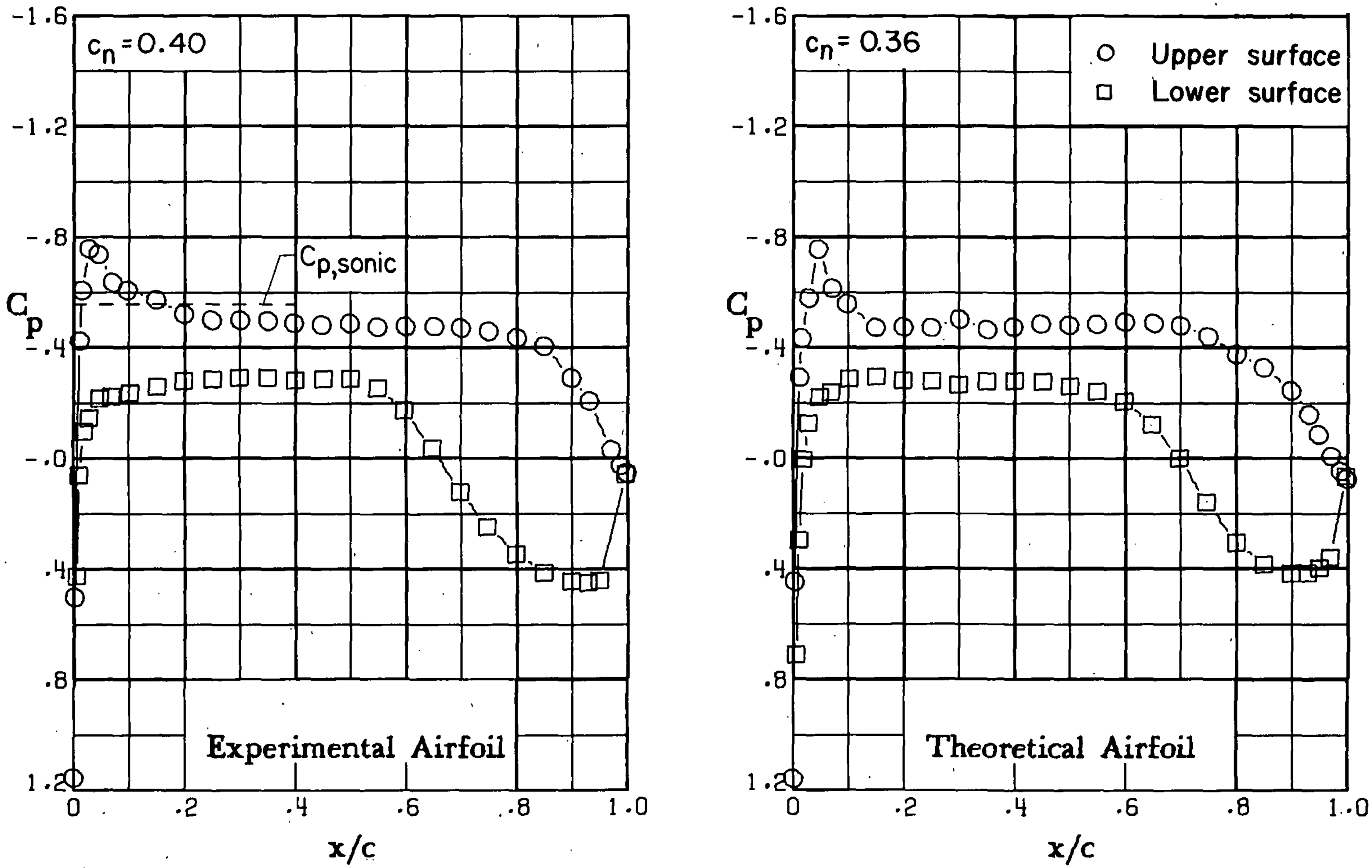
[redacted]



(b) M = 0.76; α = 0.0°.

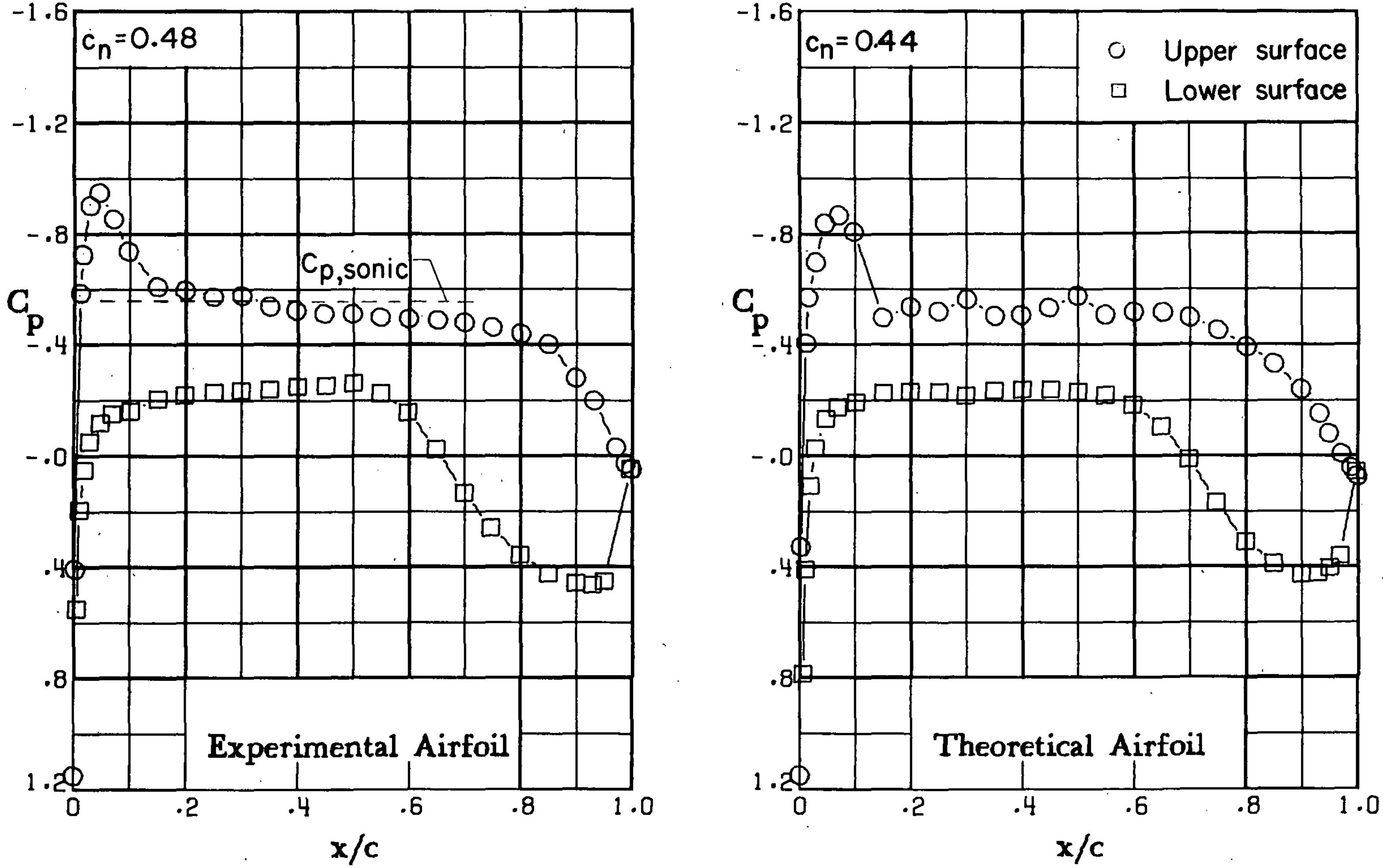
Figure 20.- Continued.

[redacted]



(c) $M = 0.76$; $\alpha = 0.5^{\circ}$.

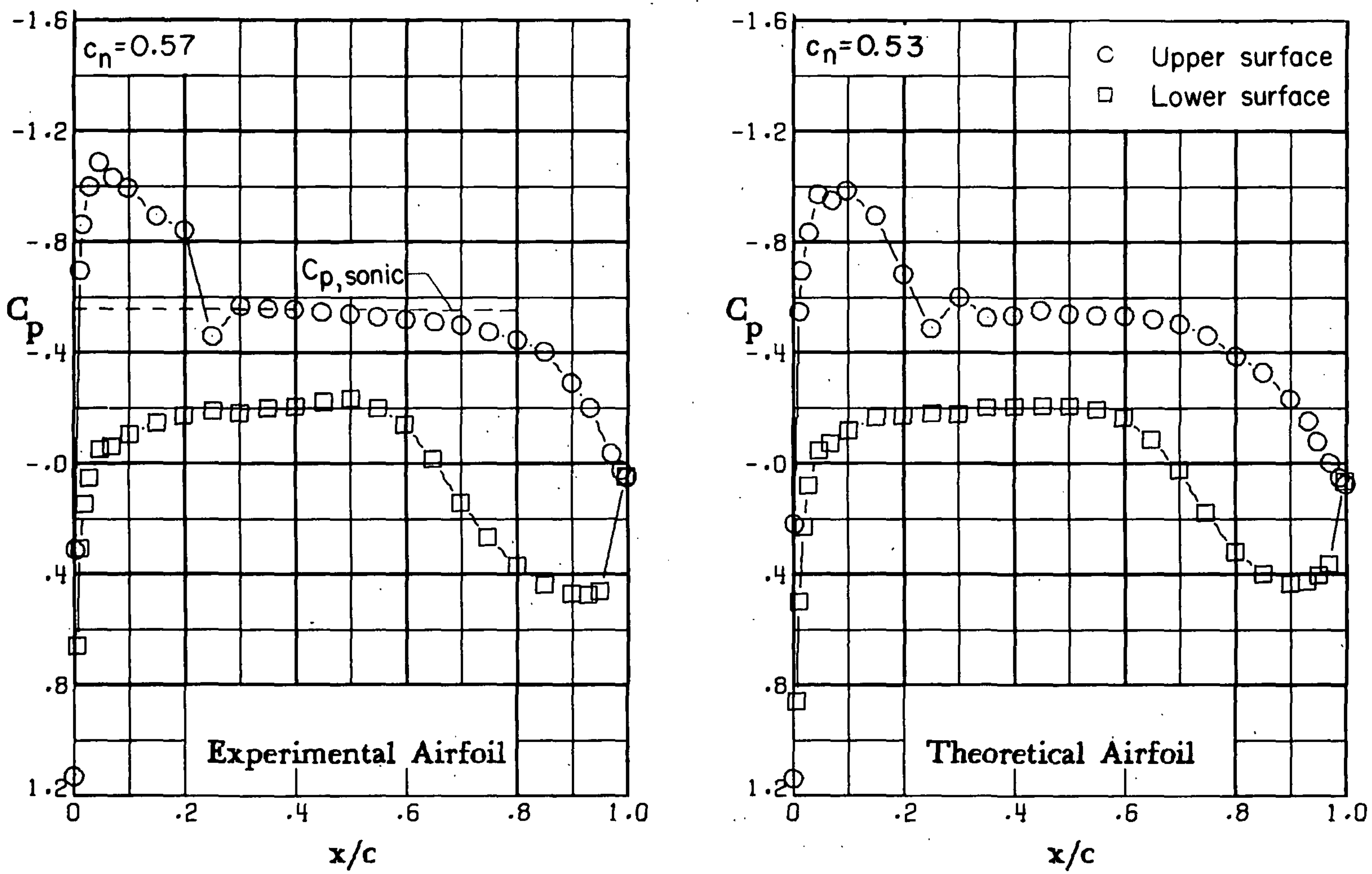
Figure 20.- Continued.



(d) M = 0.76; $\alpha = 1.0^{\circ}$.

Figure 20.- Continued.

[redacted]

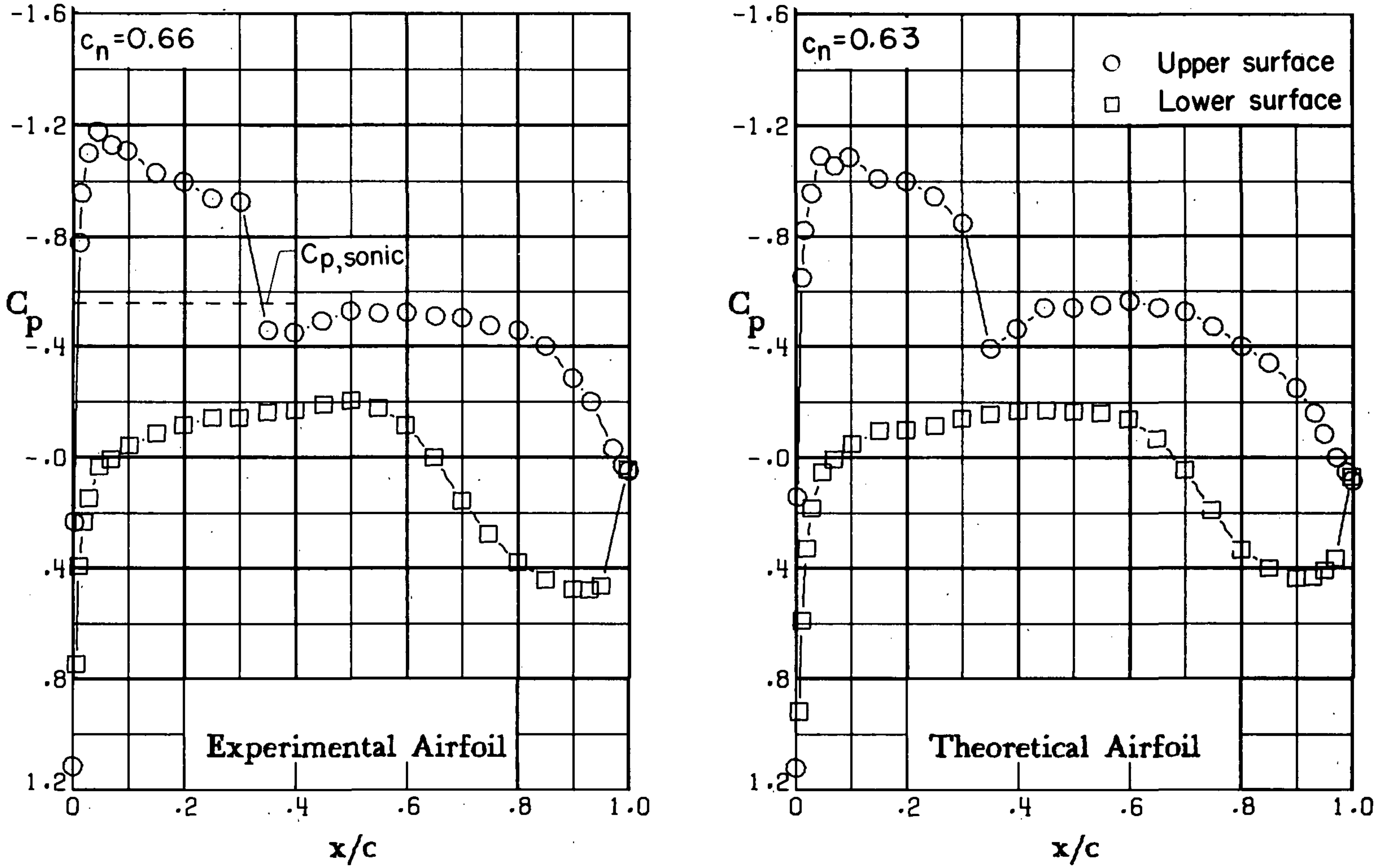


(e) $M = 0.76$; $\alpha = 1.5^{\circ}$.

Figure 20.- Continued.

[redacted]

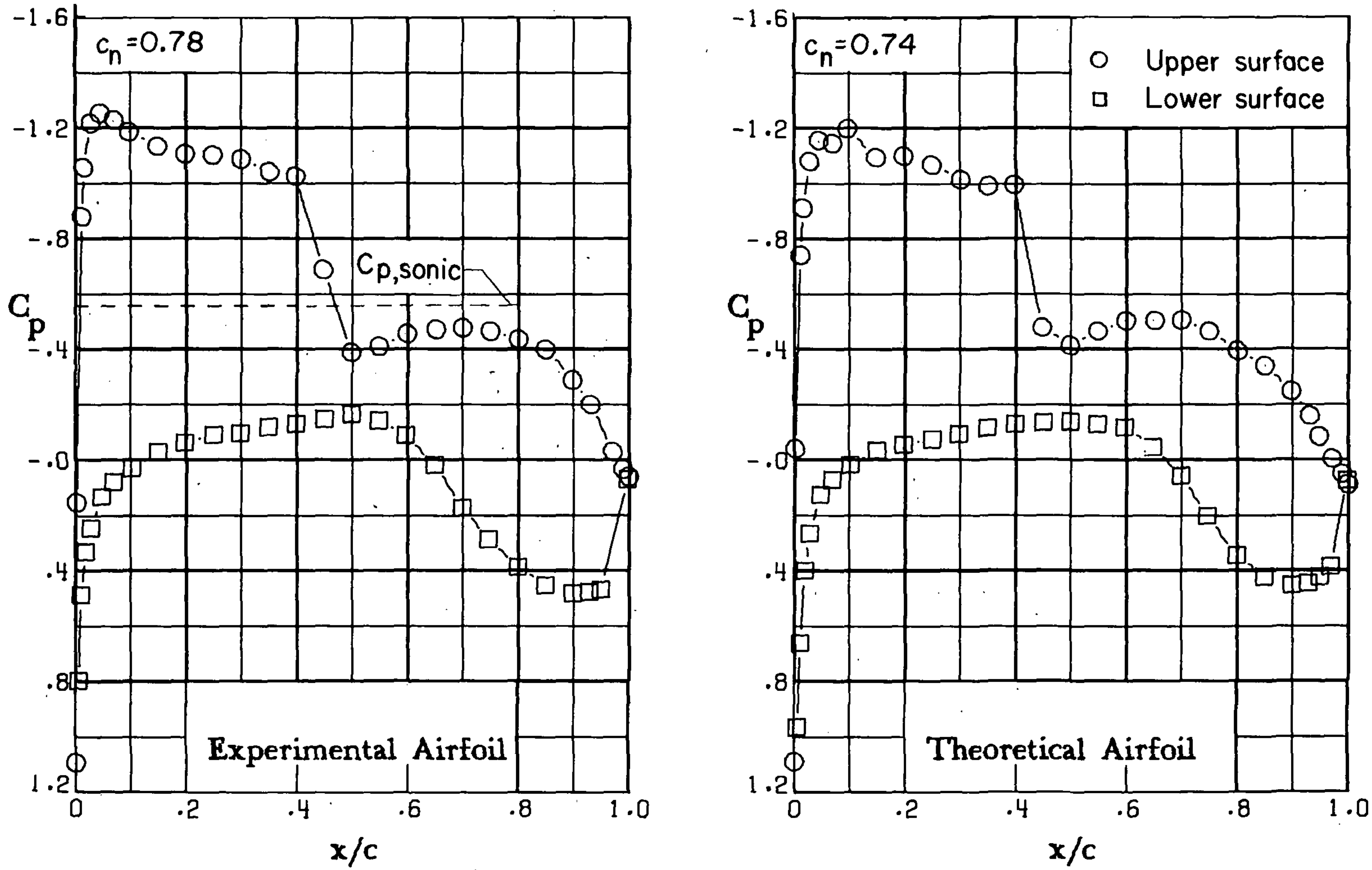
[redacted]



(f) M = 0.76; α = 2.0°.

Figure 20.- Continued.

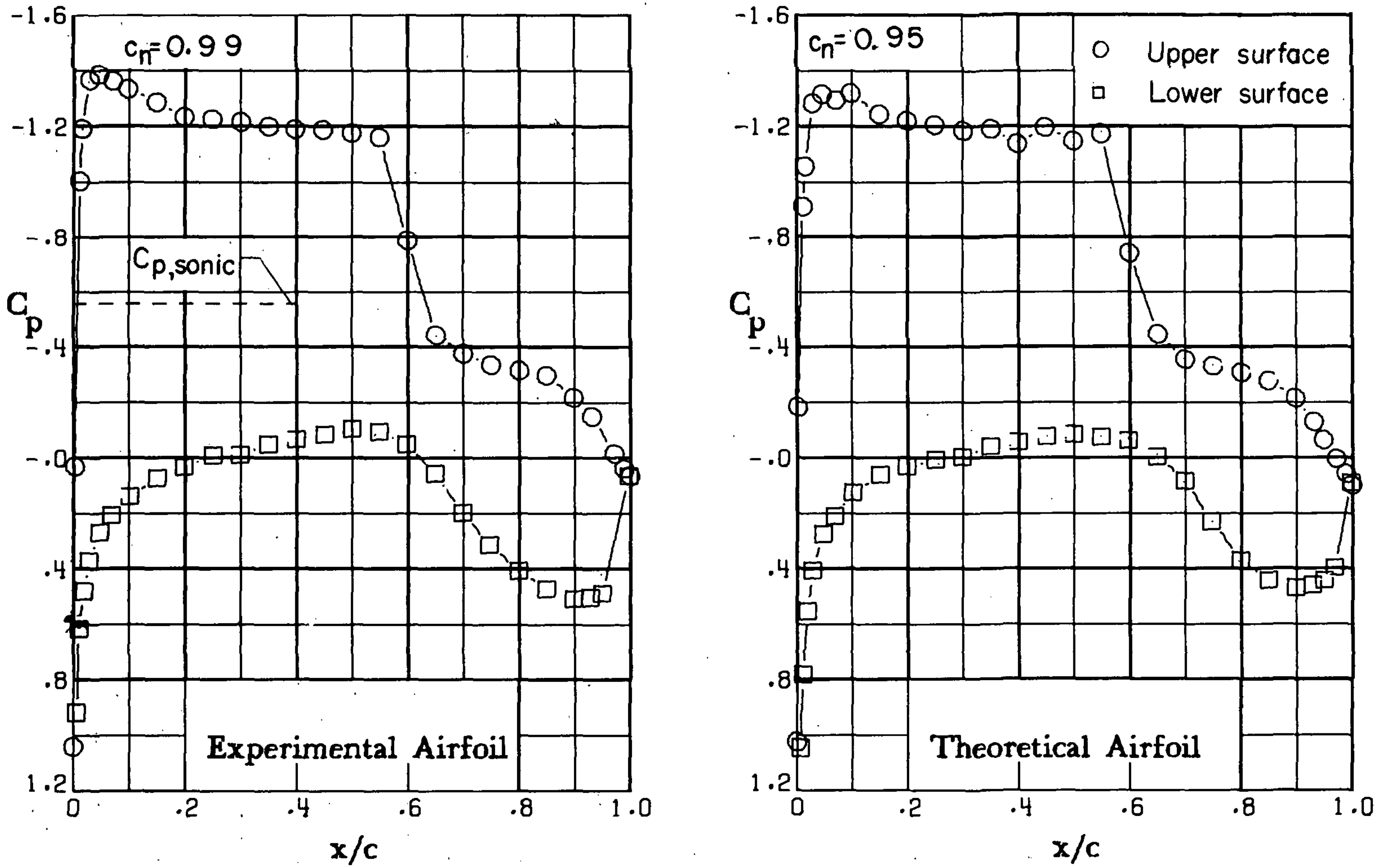
[redacted]



(g) M = 0.76; α = 2.5°.

Figure 20.- Continued.

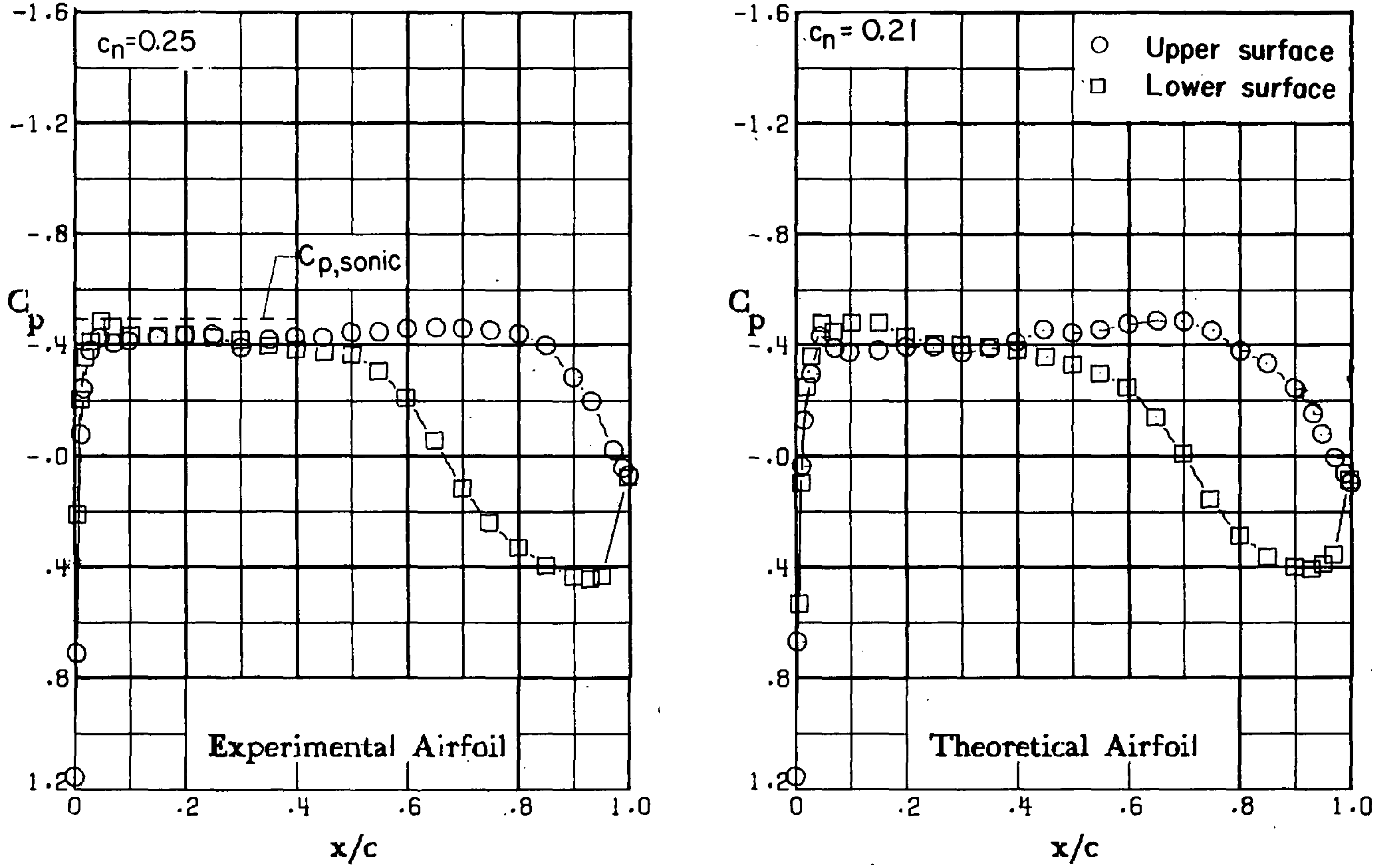
[redacted]



(h) $M = 0.76$; $\alpha = 3.5^{o}$.

Figure 20.- Concluded.

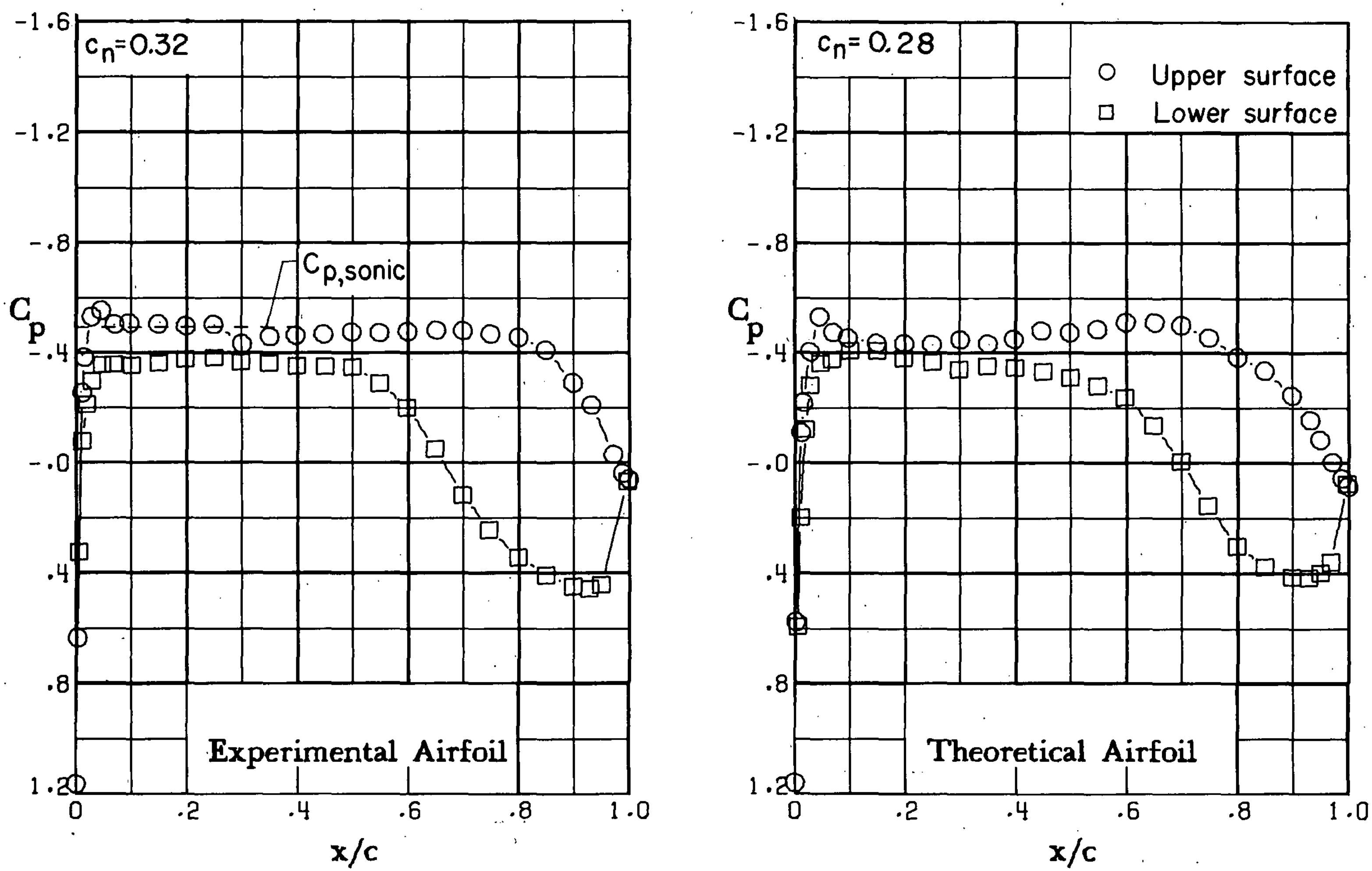
[redacted]



(a) $M = 0.78;\ \alpha = -0.5^{\circ}$.

Figure 21.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

[redacted]

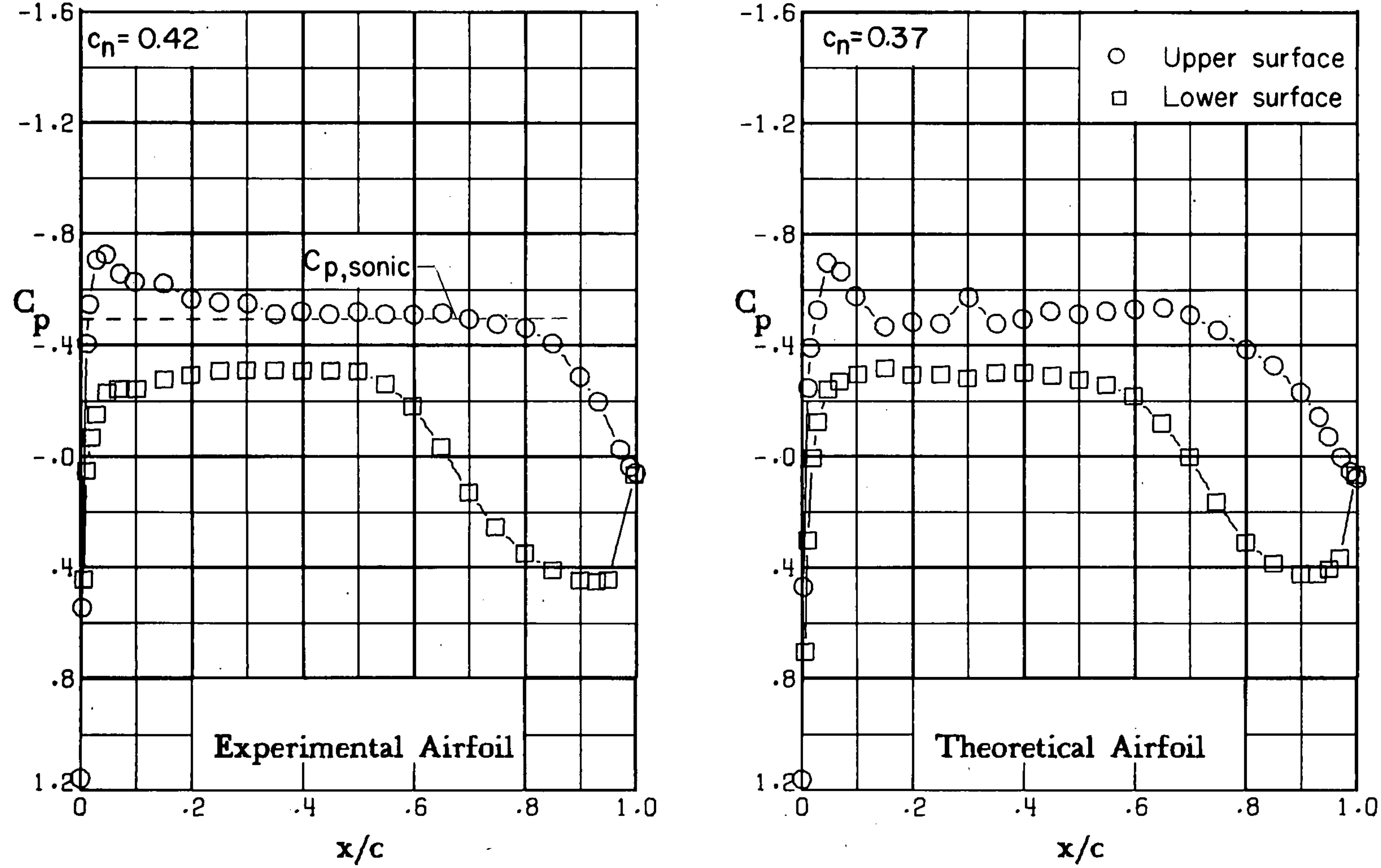


(b) M = 0.78; $\alpha = 0.0^{\circ}$.

Figure 21.- Continued.

[redacted]

[redacted]

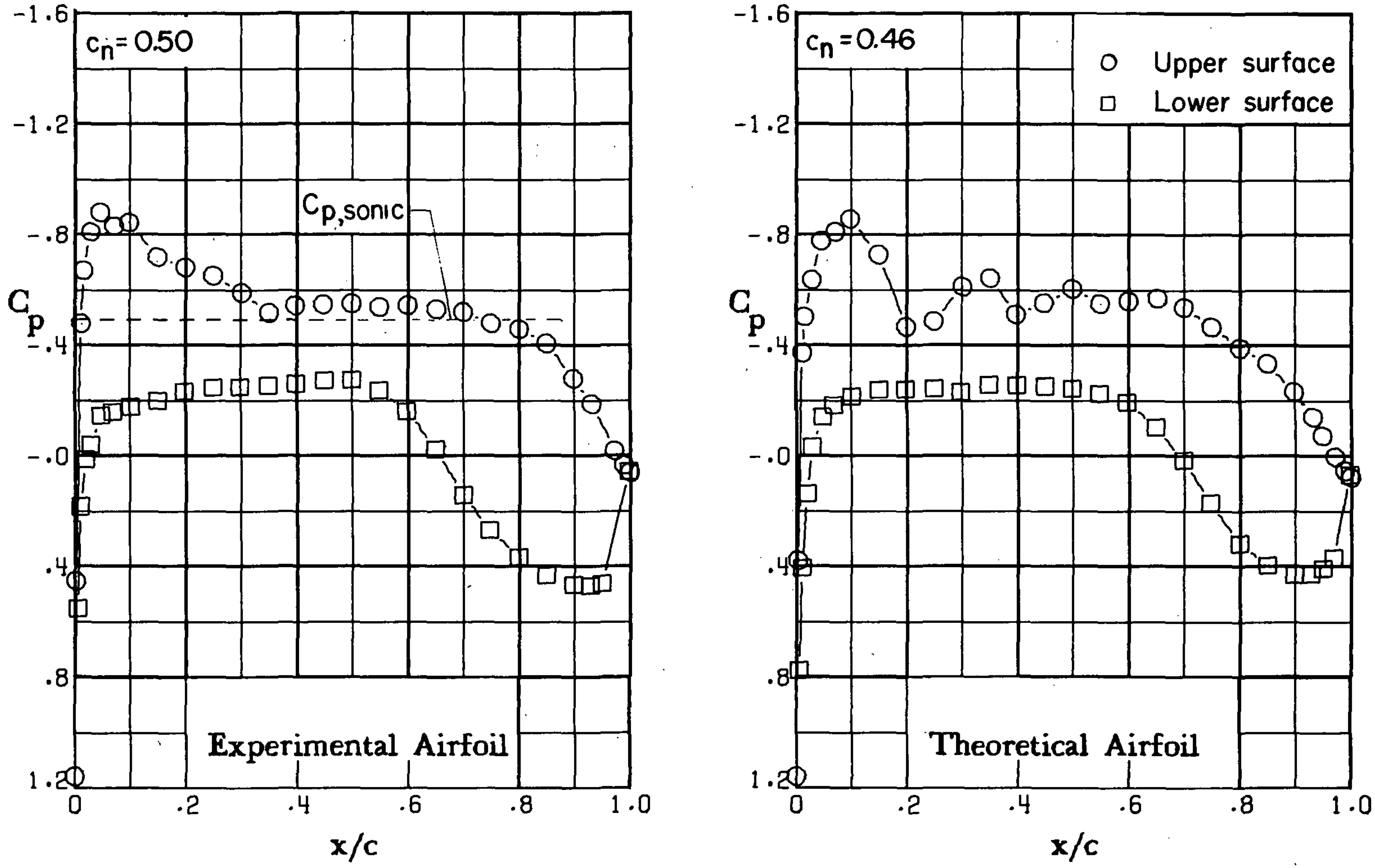


(c) M = 0.78; α = 0.5°.

Figure 21.- Continued.

[redacted]

[redacted]

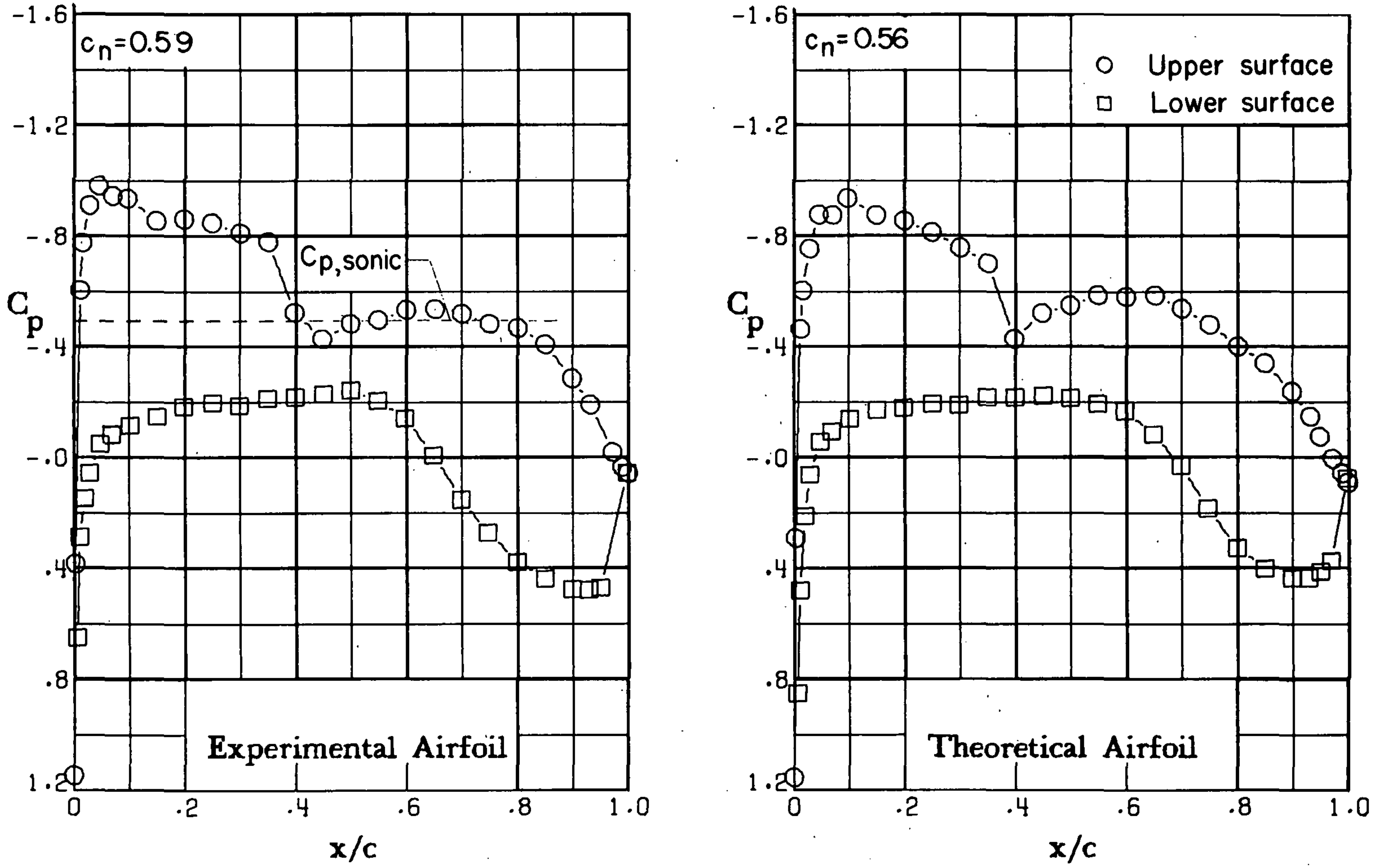


(d) M = 0.78; α = 1.0°.

Figure 21.- Continued.

[redacted]

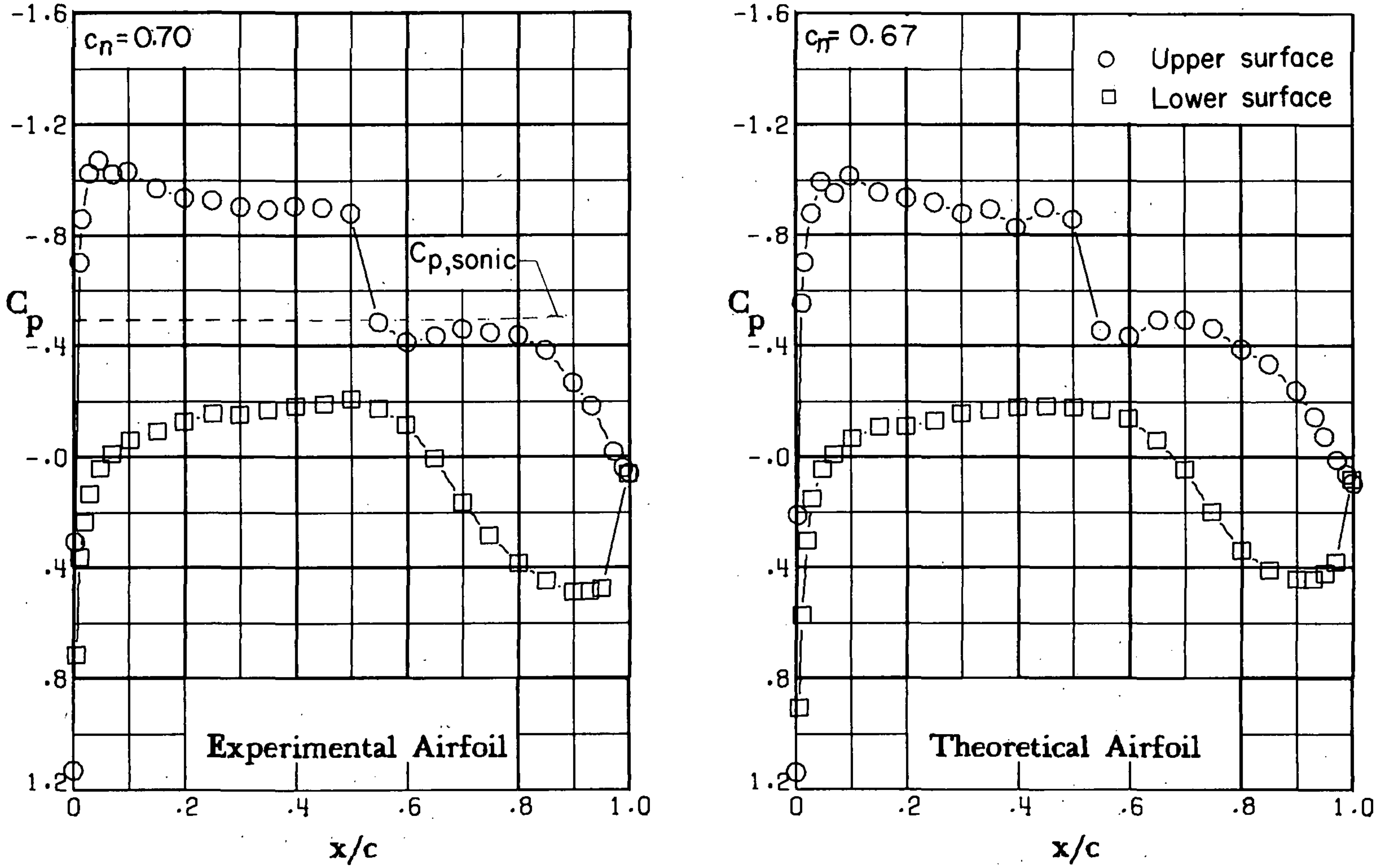
[redacted]



(e) M = 0.78; $\alpha = 1.5^{\circ}$.

Figure 21.- Continued.

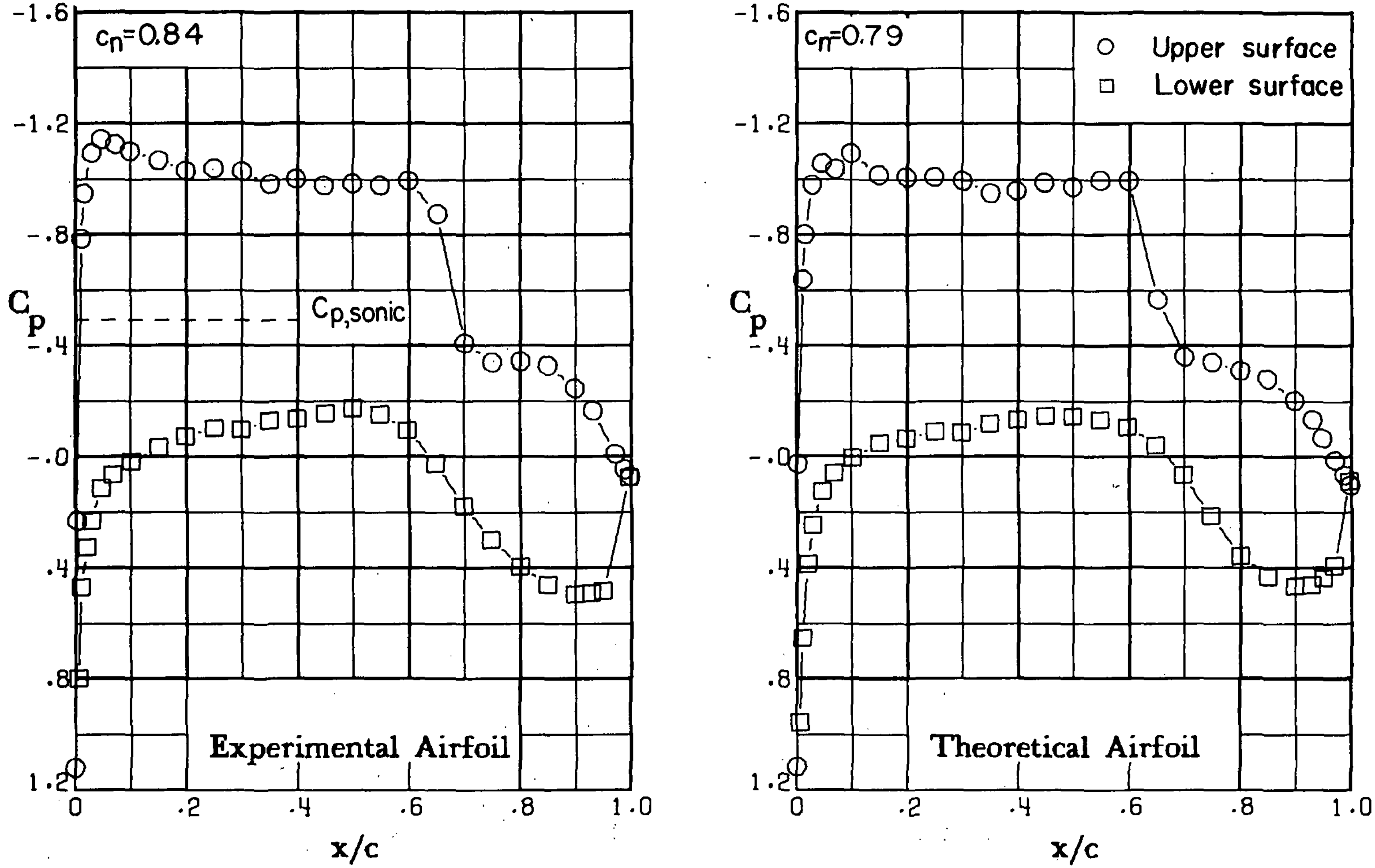
[redacted]



(f) M = 0.78; α = 2.0°.

Figure 21.- Continued.

[redacted]

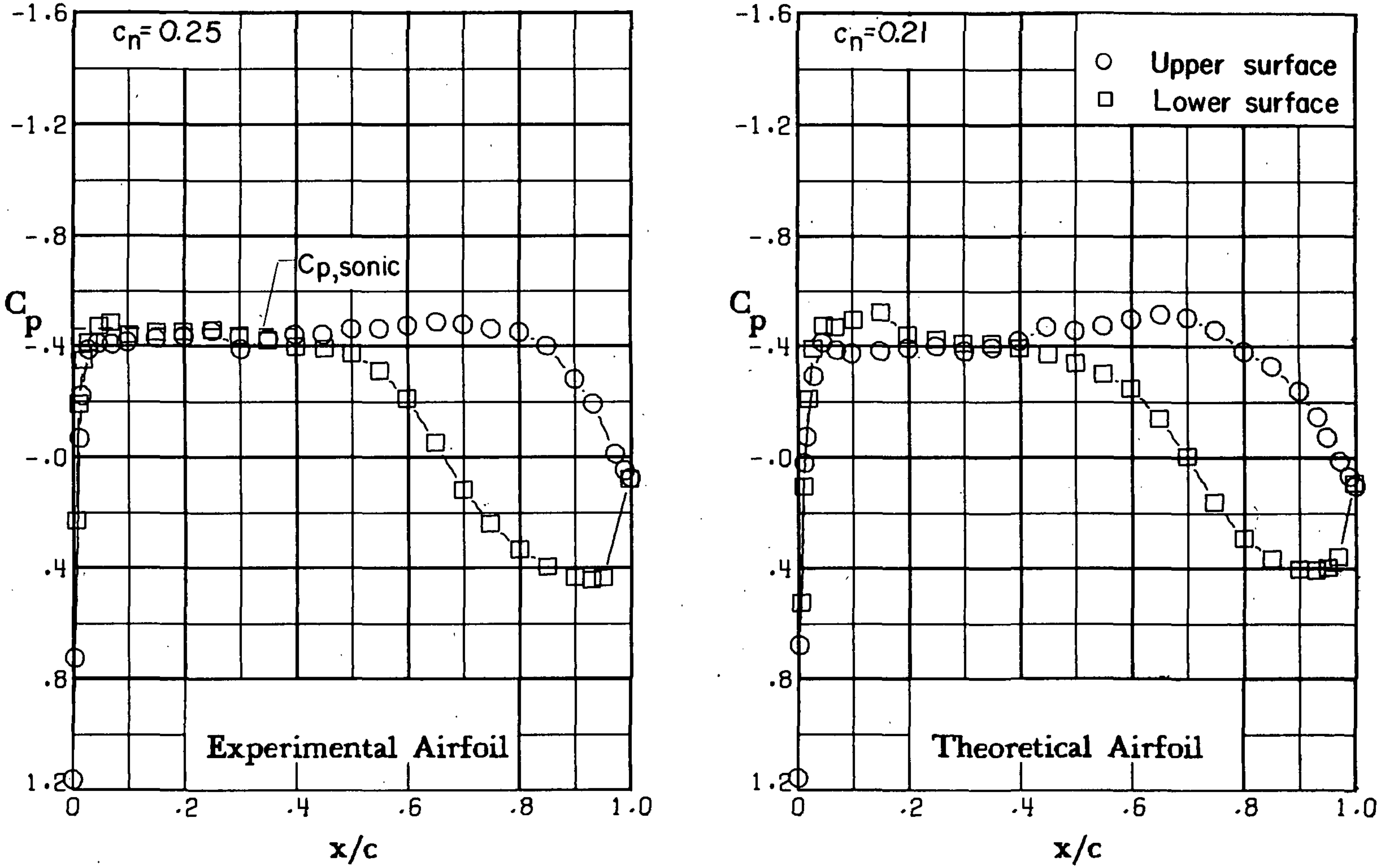


(g) M = 0.78; α = 2.5°.

Figure 21.- Concluded.

[redacted]

[redacted]

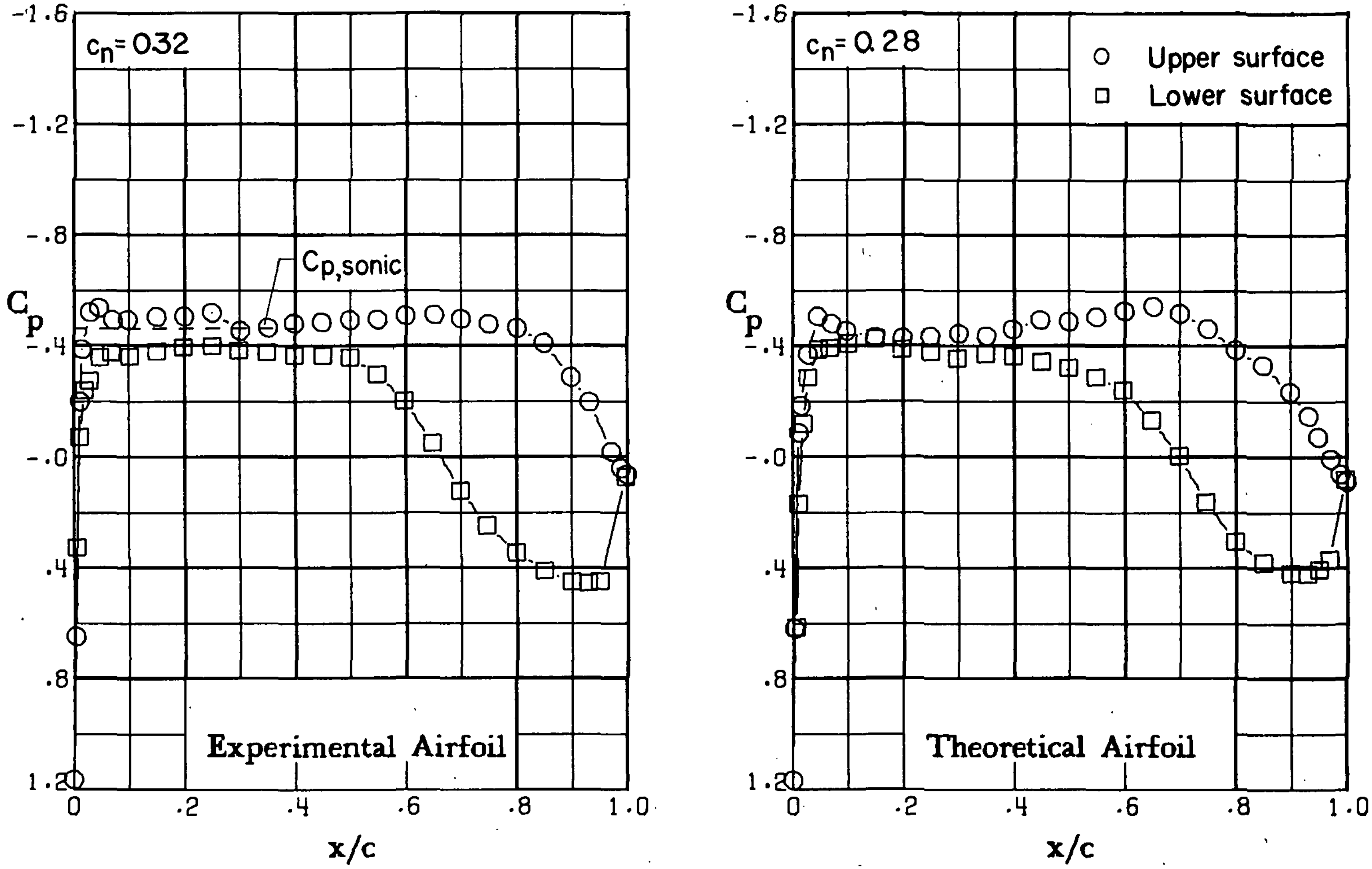


(a) $M = 0.79$; $\alpha = -0.5^{\circ}$.

Figure 22.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

[redacted]

[redacted]

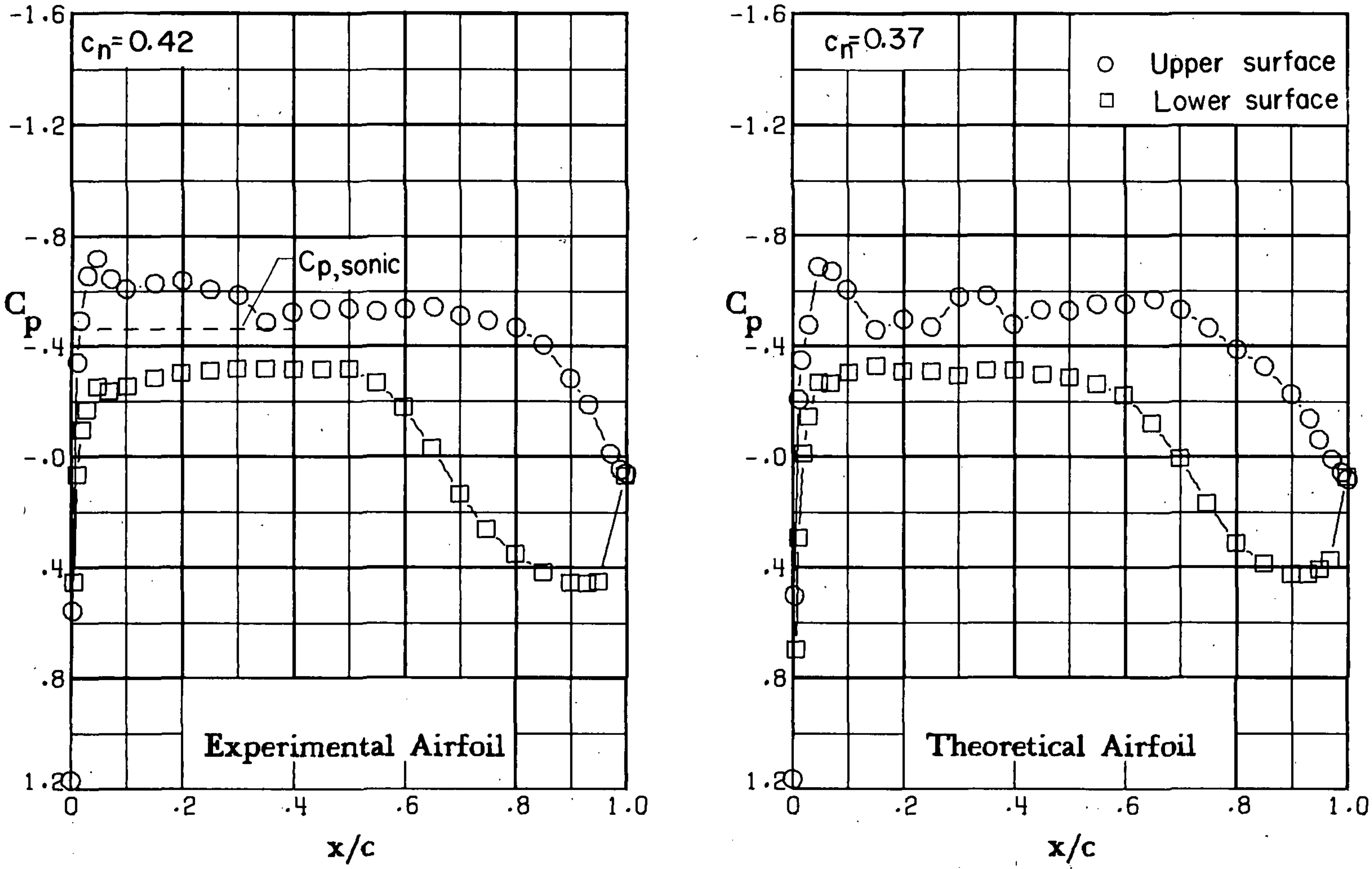


(b) M = 0.79; $\alpha = 0.0^{\circ}$.

Figure 22.- Continued.

[redacted]

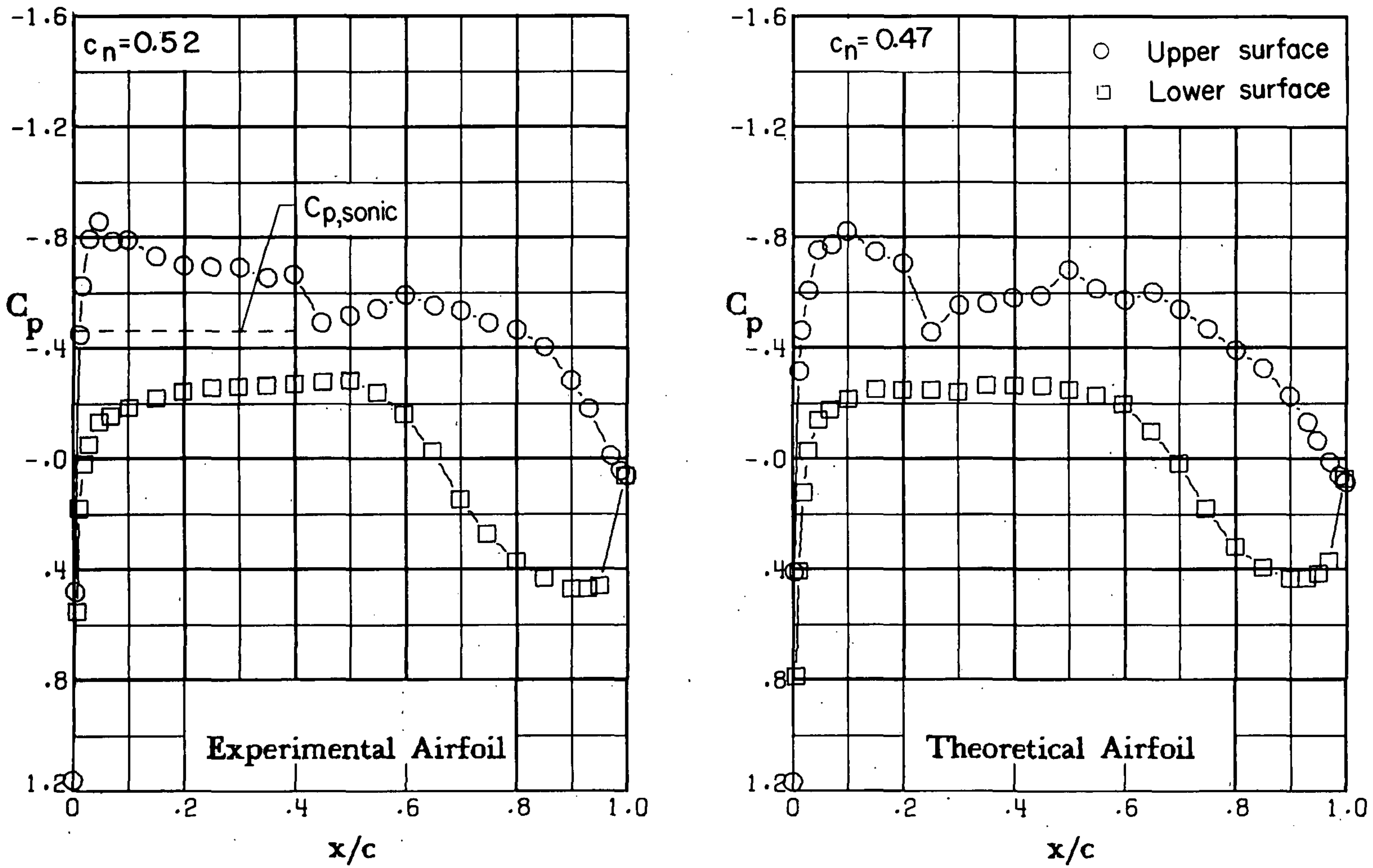
[redacted]



(c) M = 0.79; α = 0.5°.

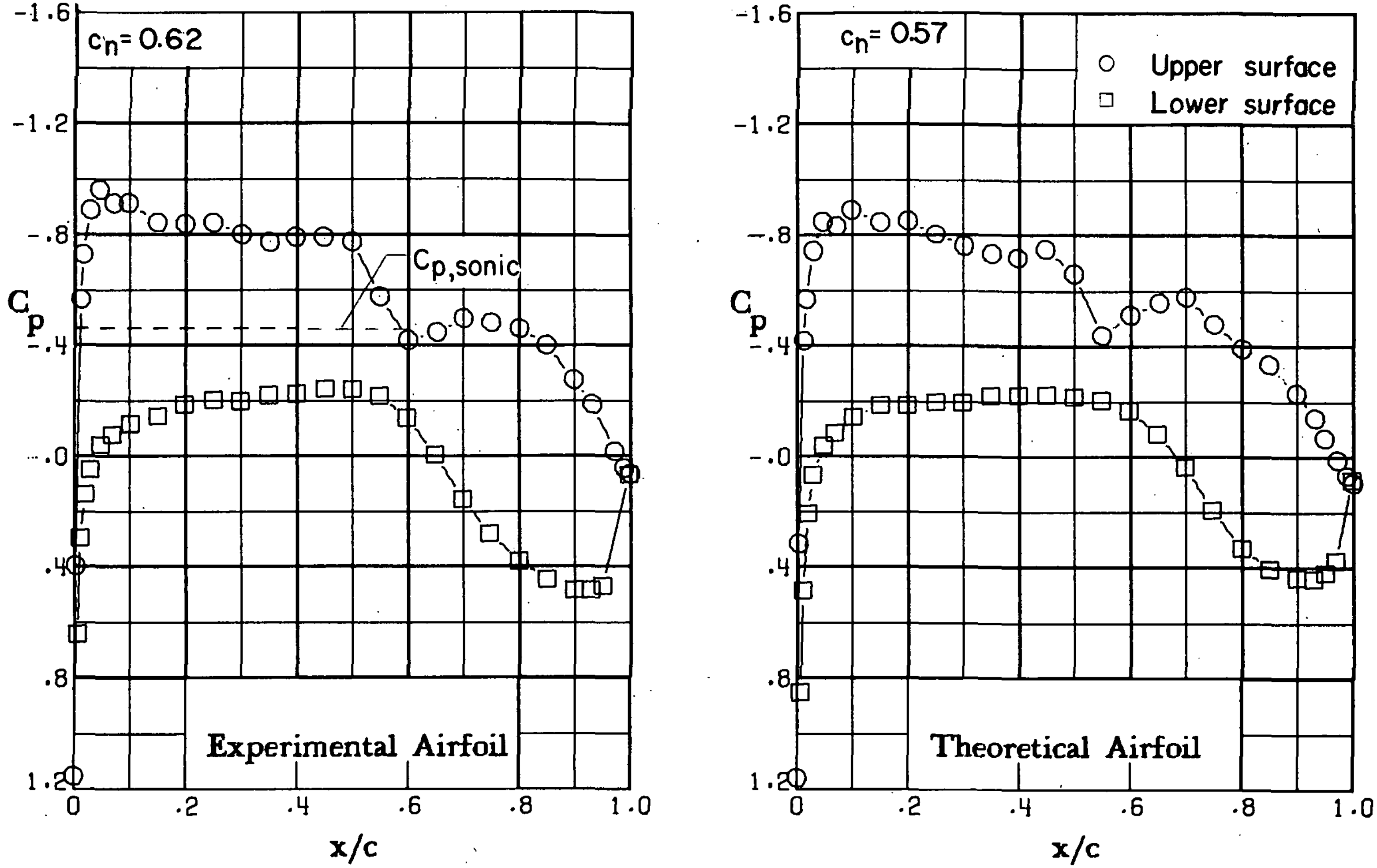
Figure 22.- Continued.

[redacted]



(d) $M = 0.79$; $\alpha = 1.0^o$.

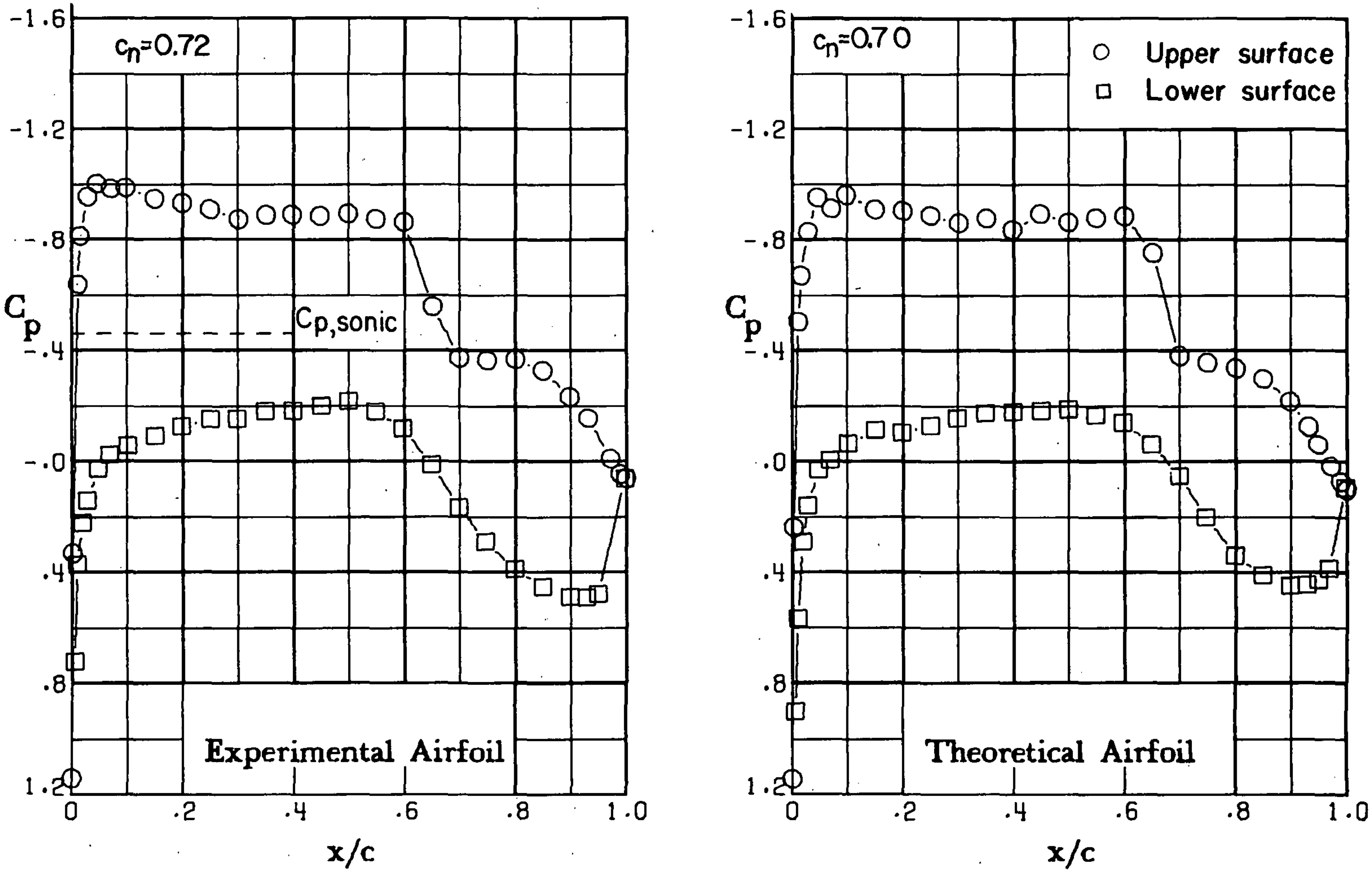
Figure 22.- Continued.



(e) $M = 0.79$; $\alpha = 1.5^{o}$.

Figure 22.- Continued.

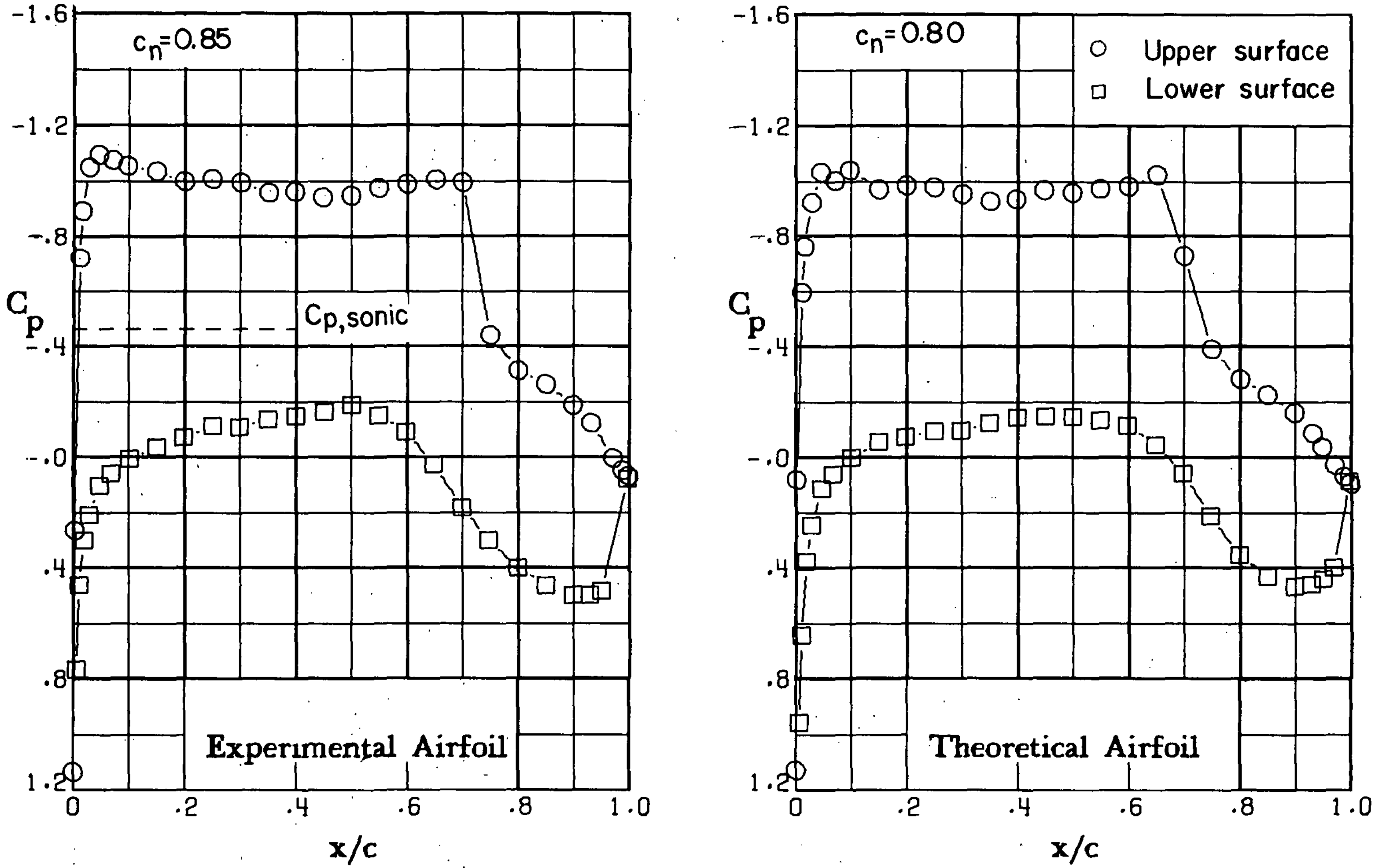
[redacted]



(f) M = 0.79; $\alpha = 2.0^{\circ}$.

Figure 22.- Continued.

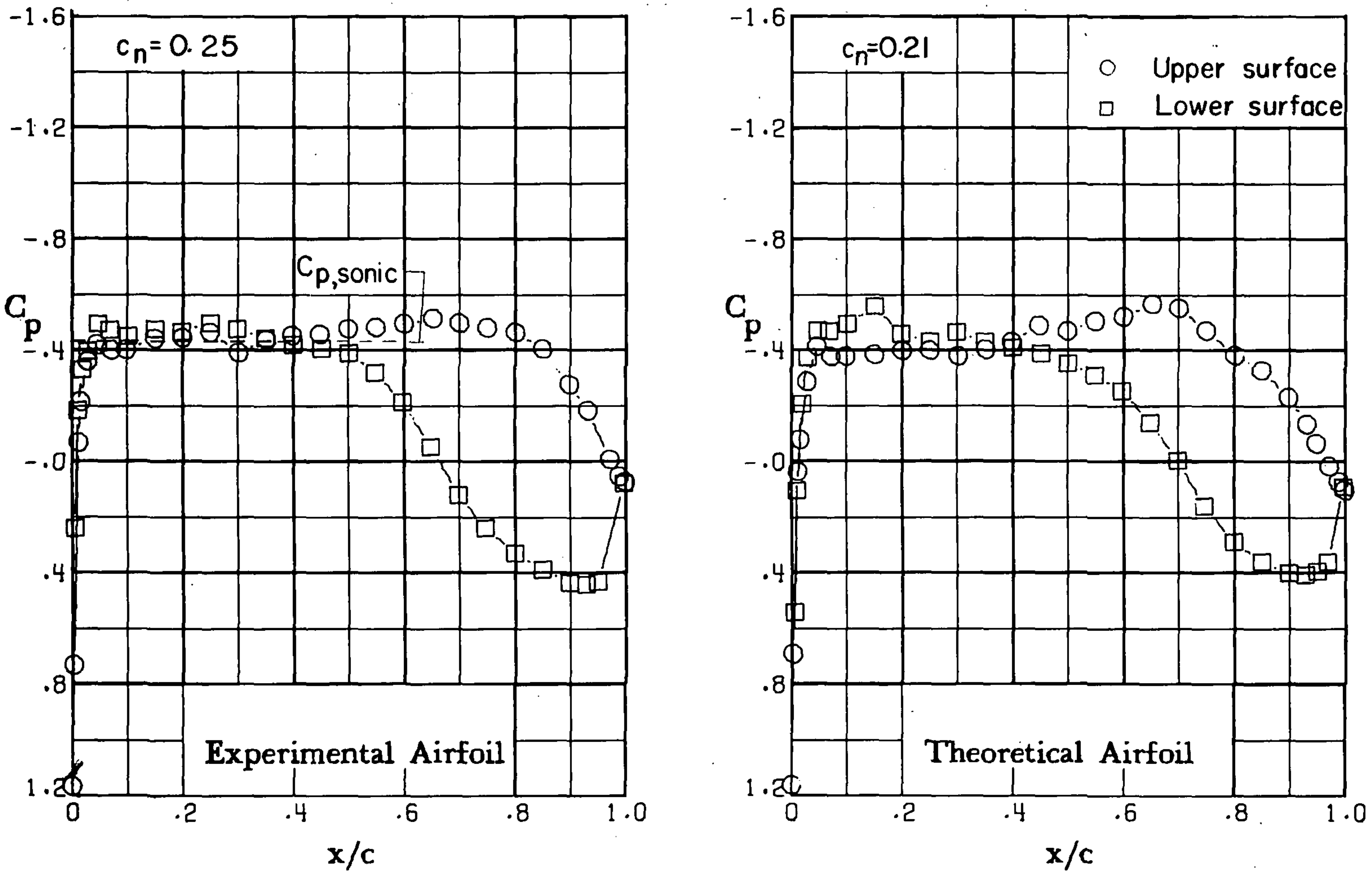
[redacted]



(g) M = 0.79; α = 2.5°.

Figure 22.- Concluded.

[redacted]

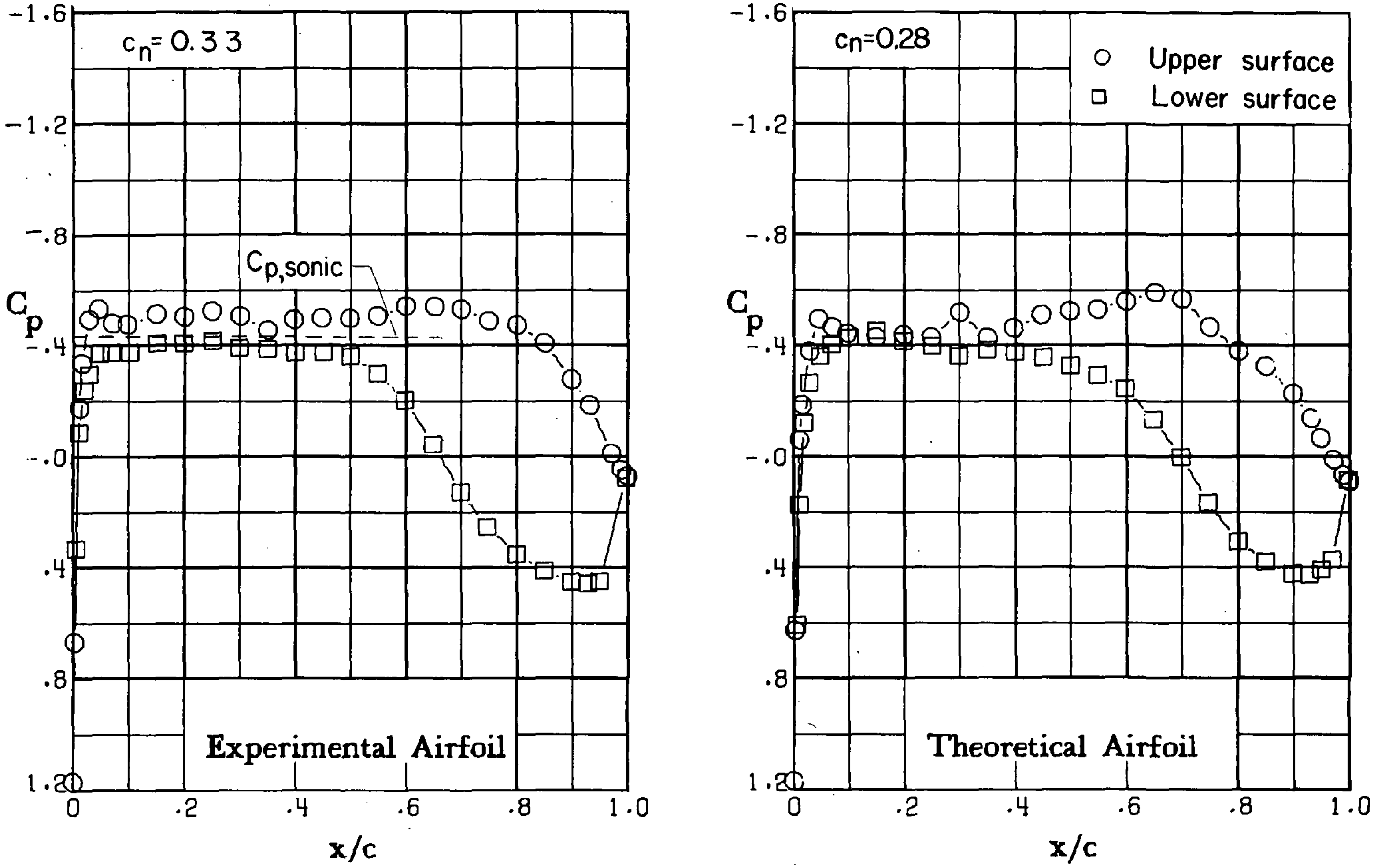


(a) M = 0.80; α = -0.5°.

Figure 23.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.

[redacted]

[redacted]

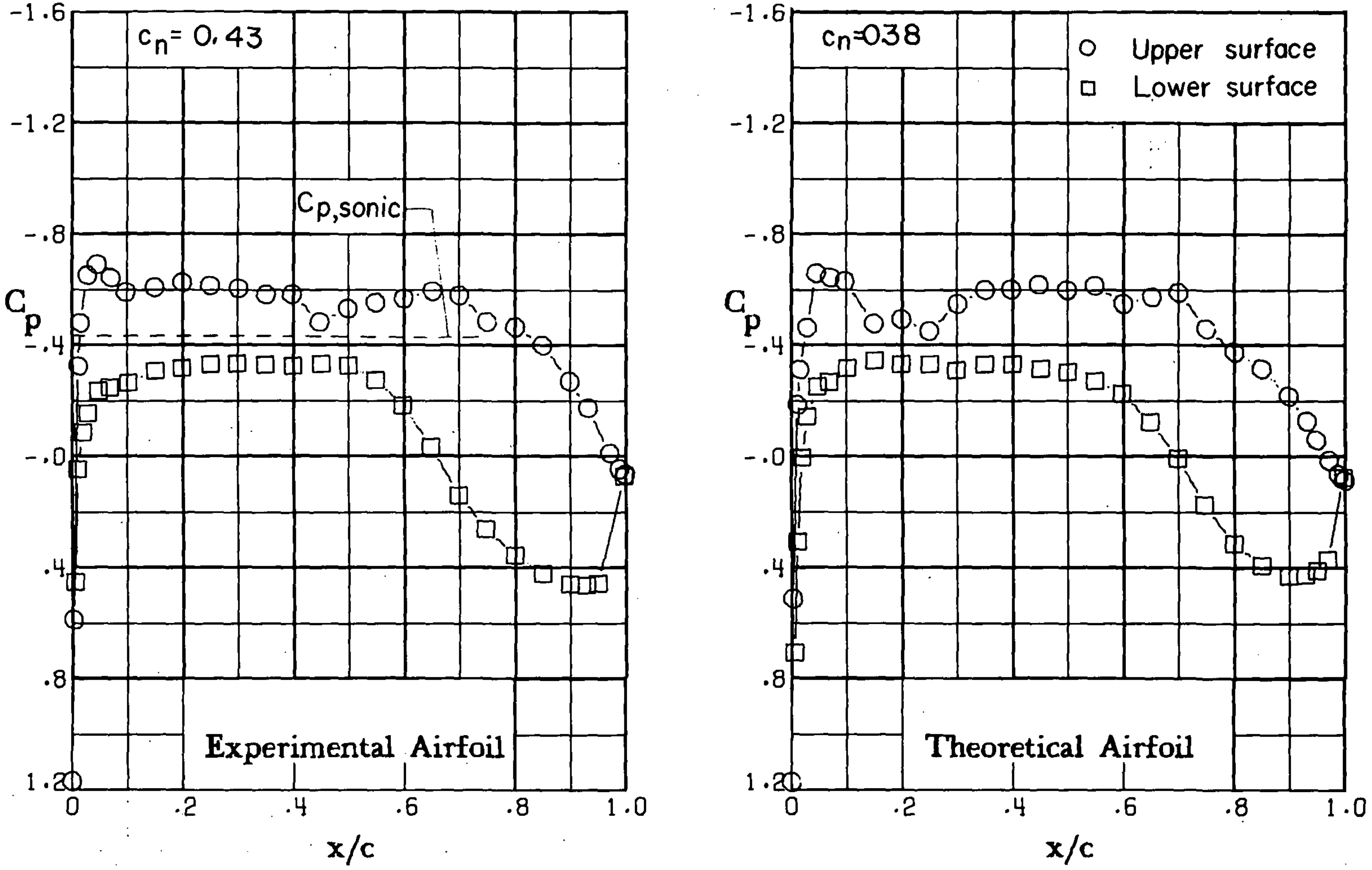


(b) M = 0.80; α = 0.0°.

Figure 23.- Continued.

[redacted]

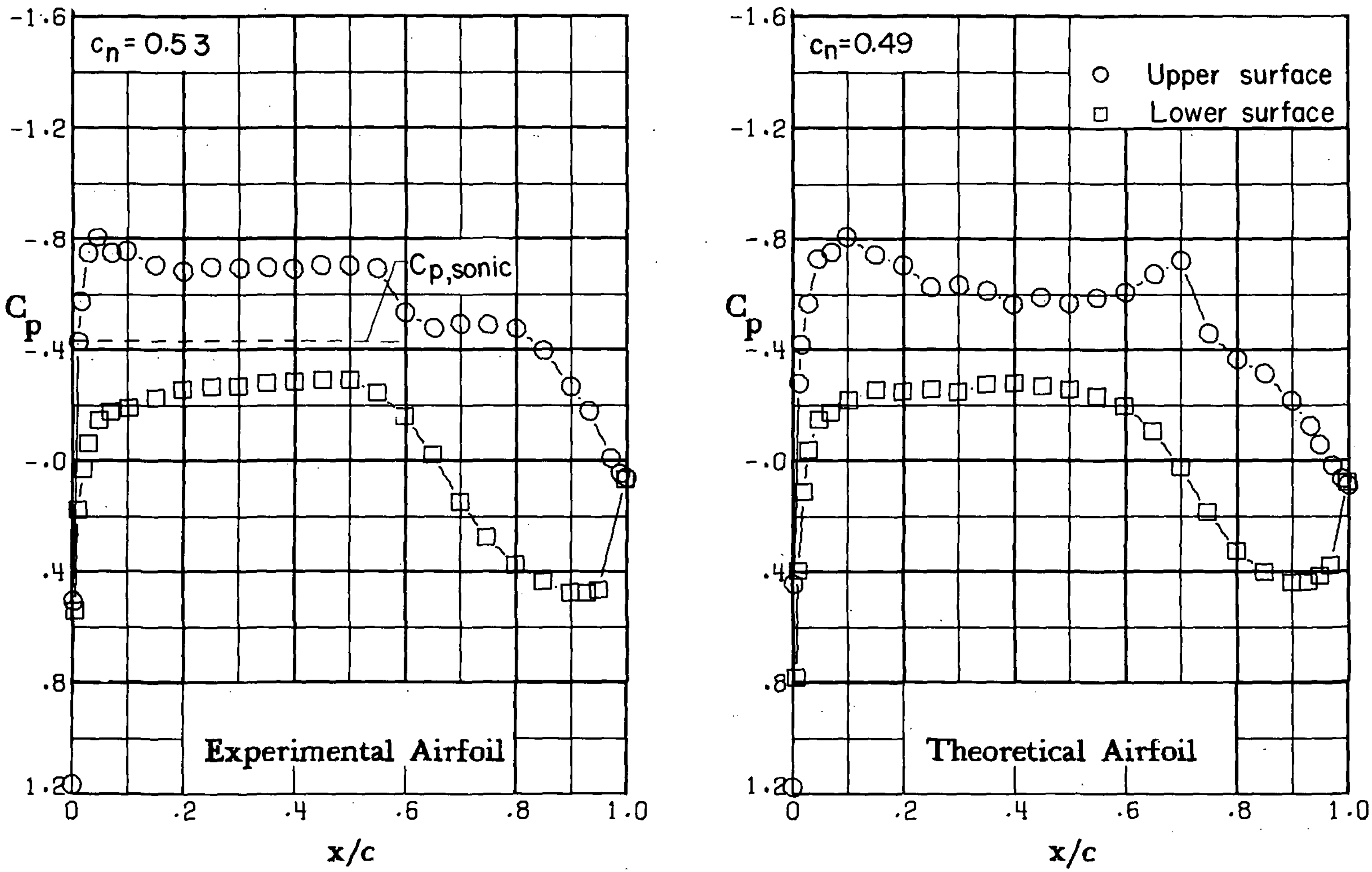
[redacted]



(c) $M = 0.80$; $\alpha = 0.5^{\circ}$.

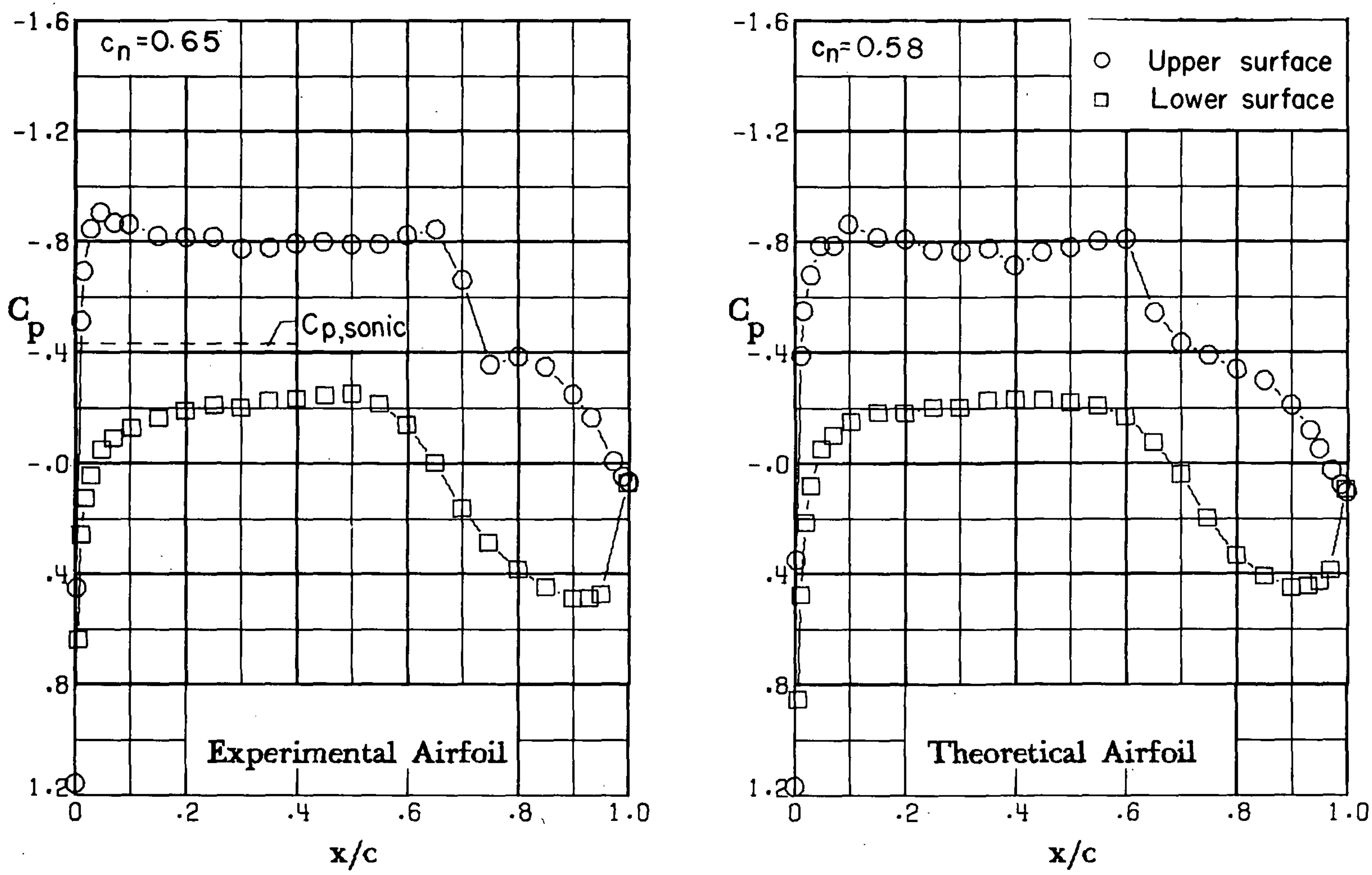
Figure 23.- Continued.

[redacted]



(d) $M = 0.80$; $\alpha = 1.0^{\circ}$.

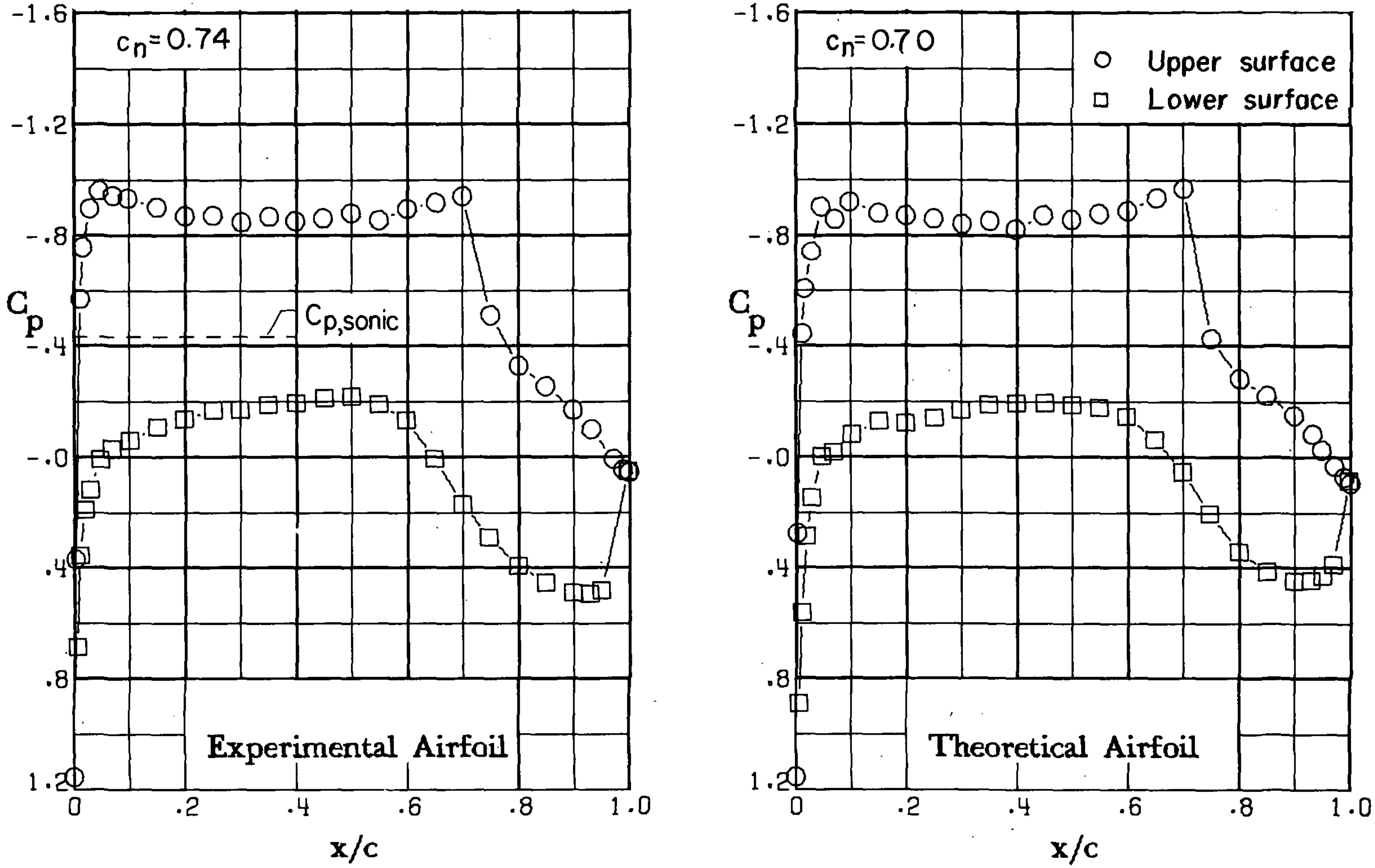
Figure 23.- Continued.



(e) $M = 0.80$; $\alpha = 1.5^{\circ}$.

Figure 23.- Continued.

[redacted]

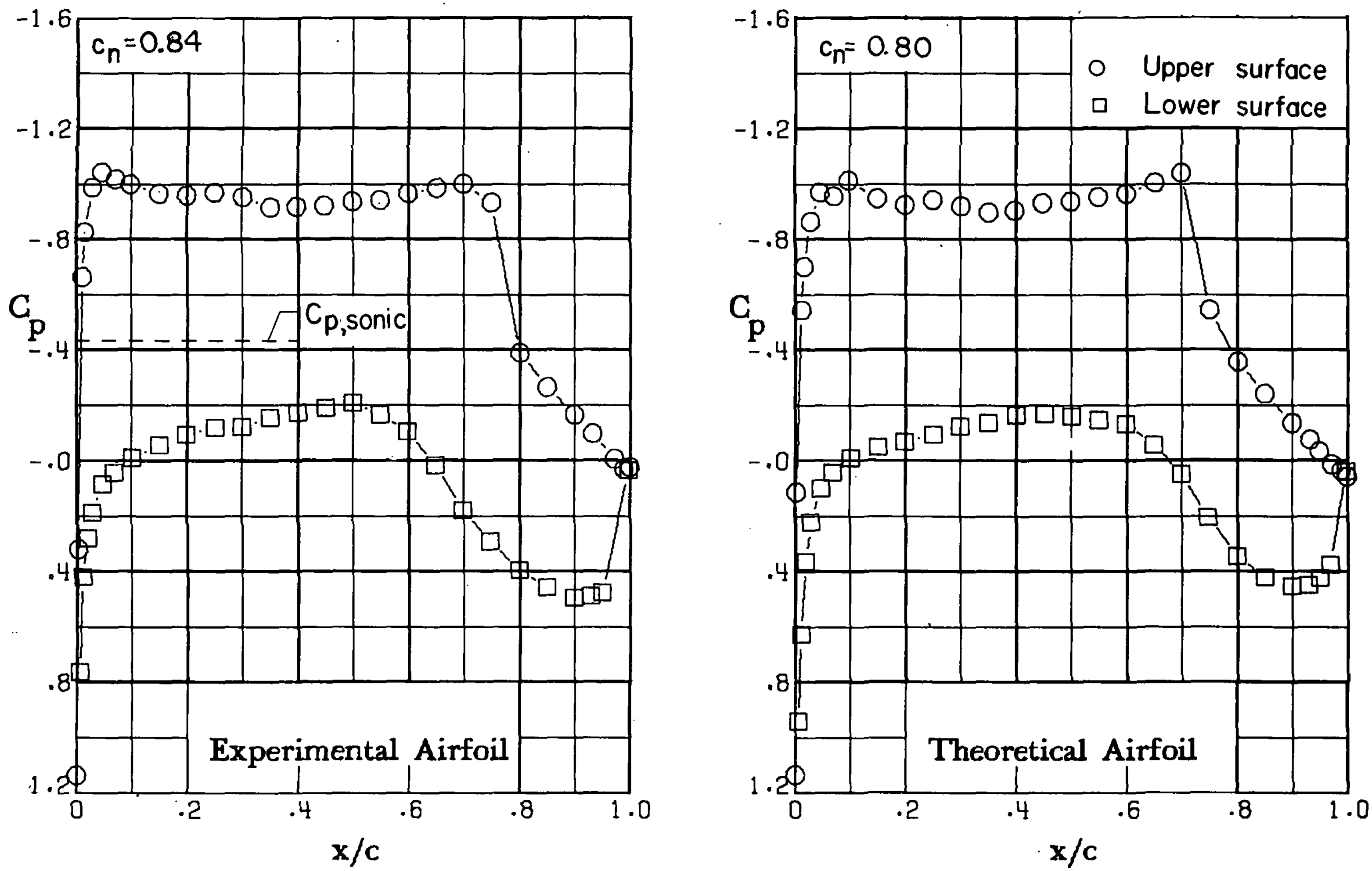


(f) M = 0.80; $\alpha = 2.0^{o}$.

Figure 23.- Continued.

[redacted]

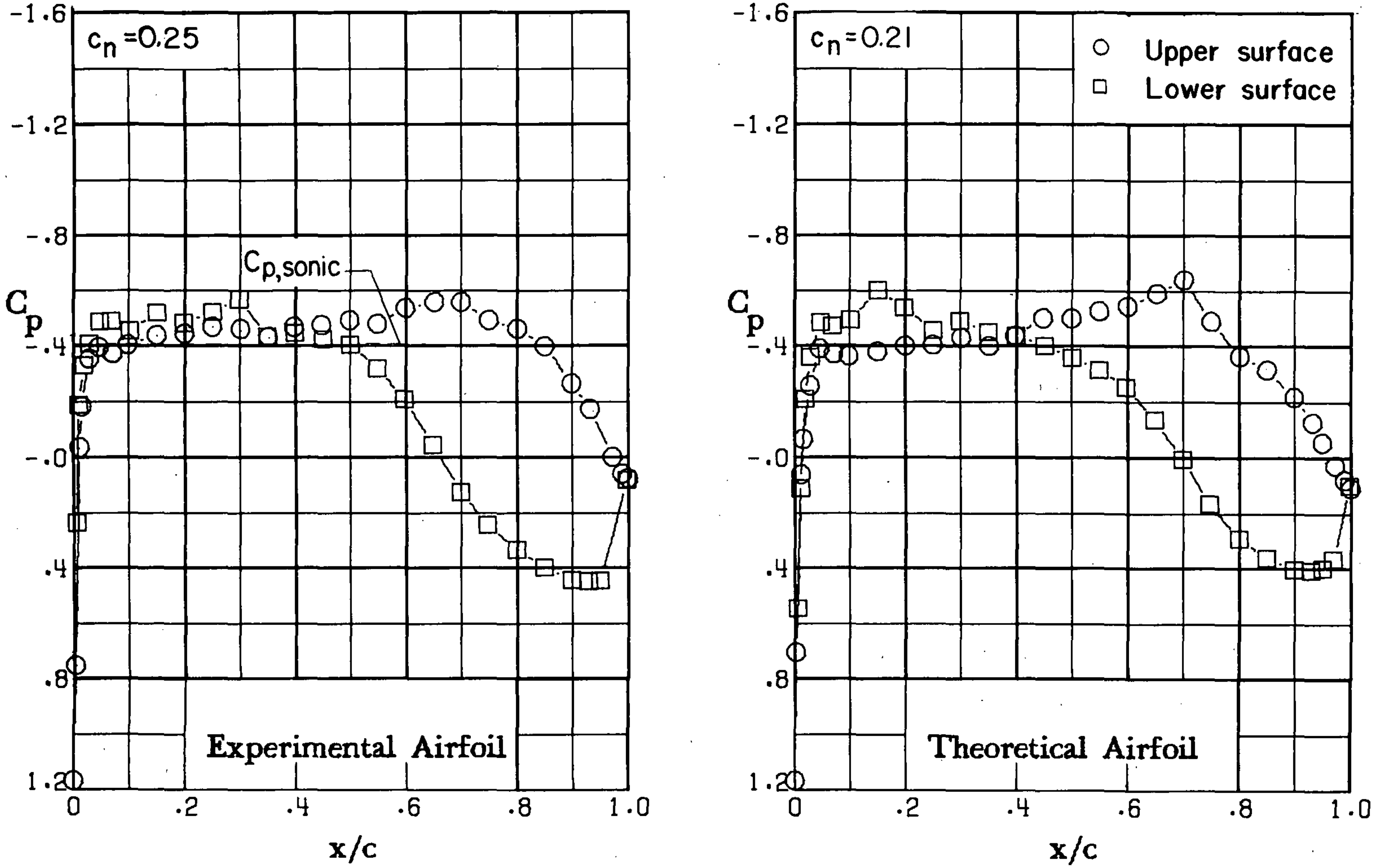
[redacted]



(g) M = 0.80; α = 2.5°.

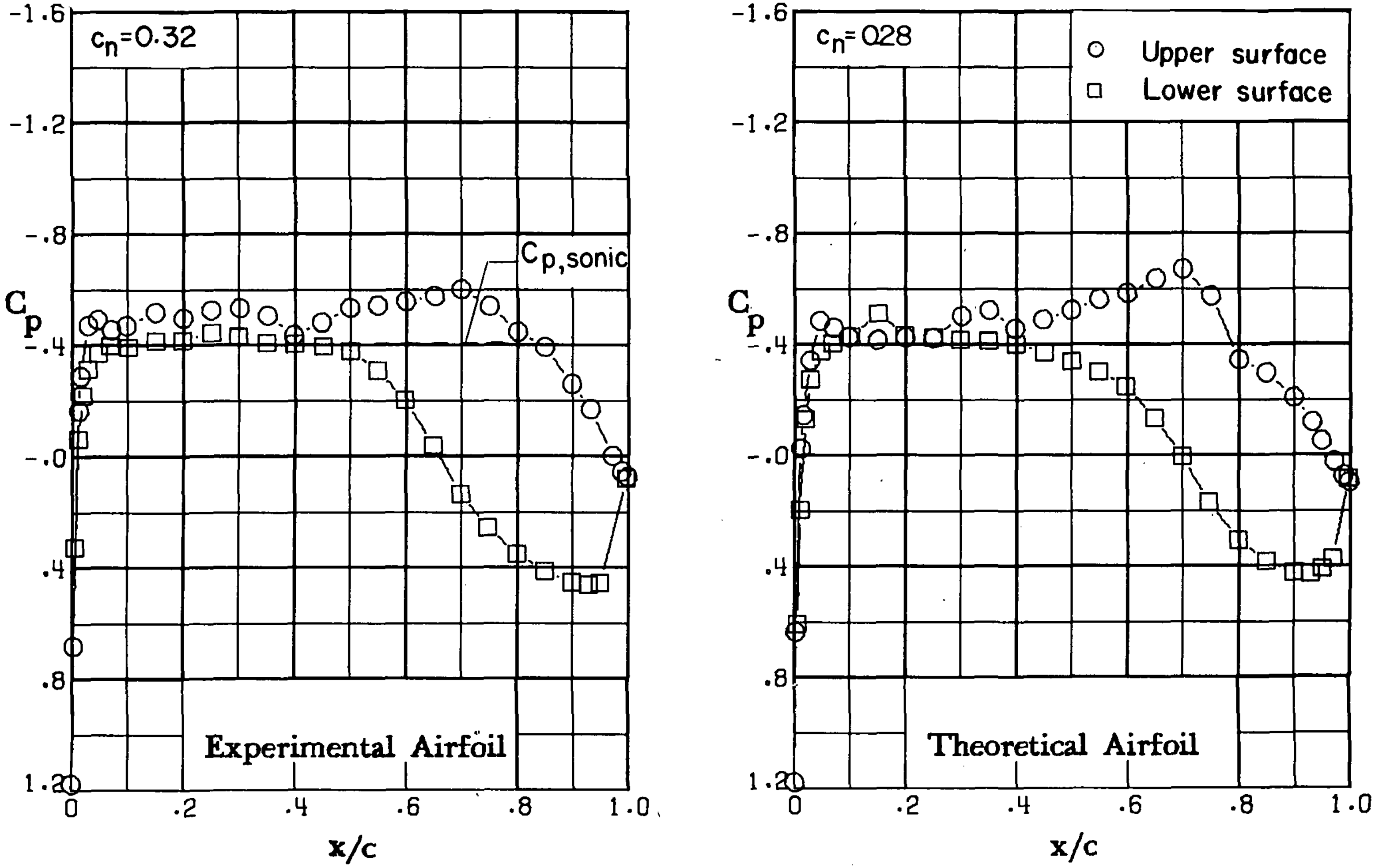
Figure 23.- Concluded.

[redacted]



(a) $M = 0.81$; $\alpha = -0.5^{o}$.

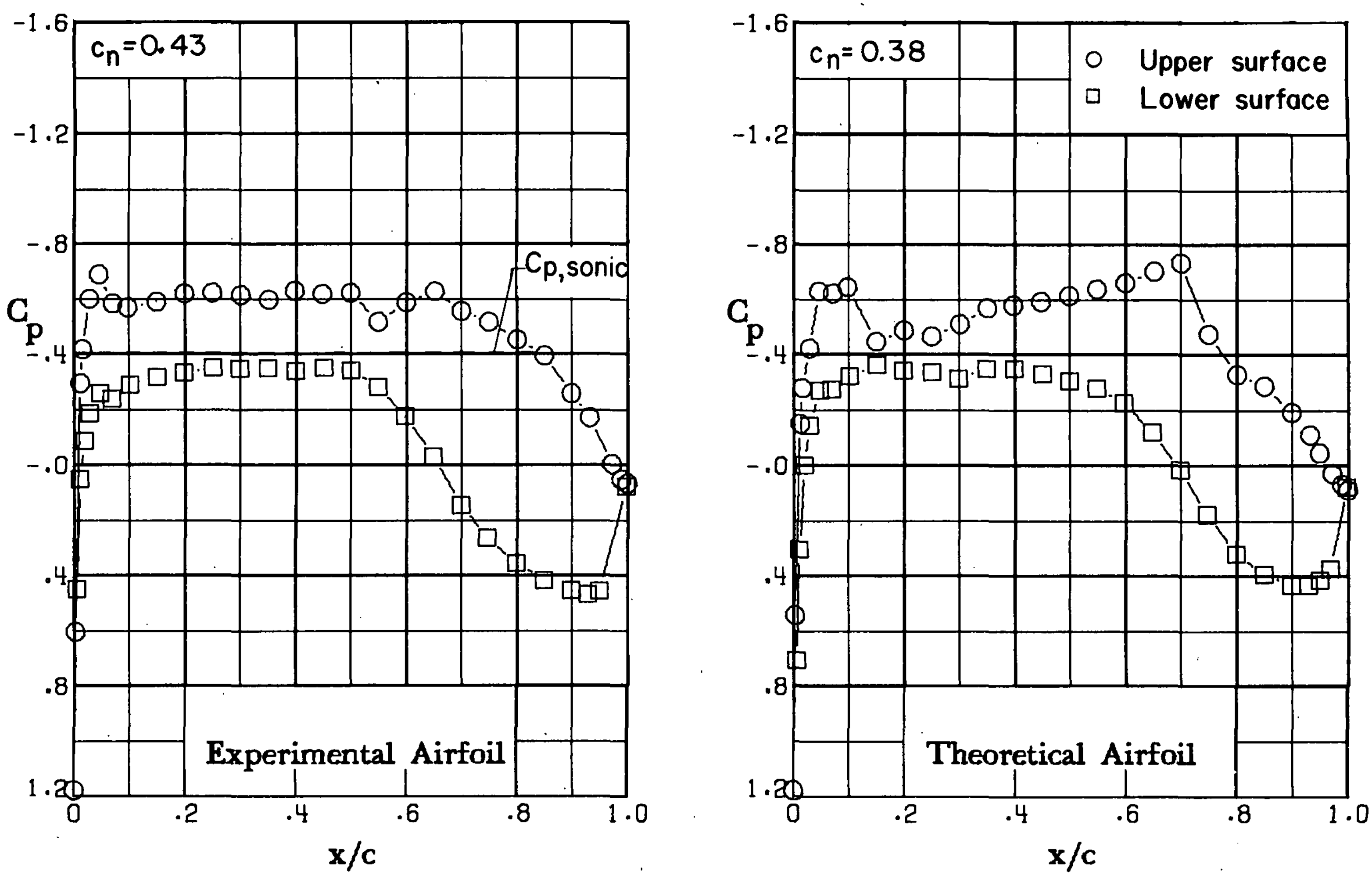
Figure 24.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.



(b) M = 0.81; α = 0.0°.

Figure 24.- Continued.

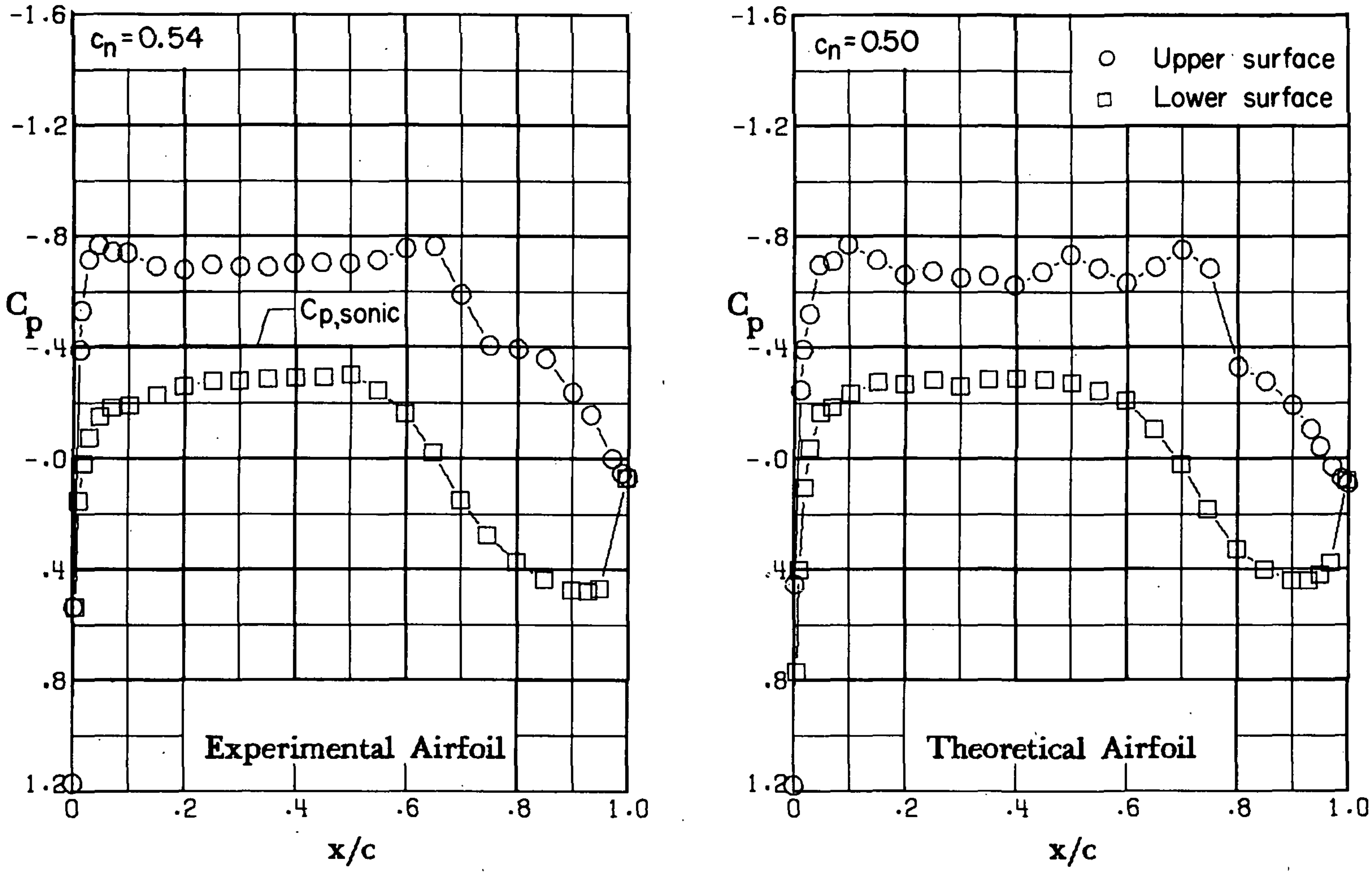
[redacted]



(c) M = 0.81; $\alpha = 0.5^{\circ}$.

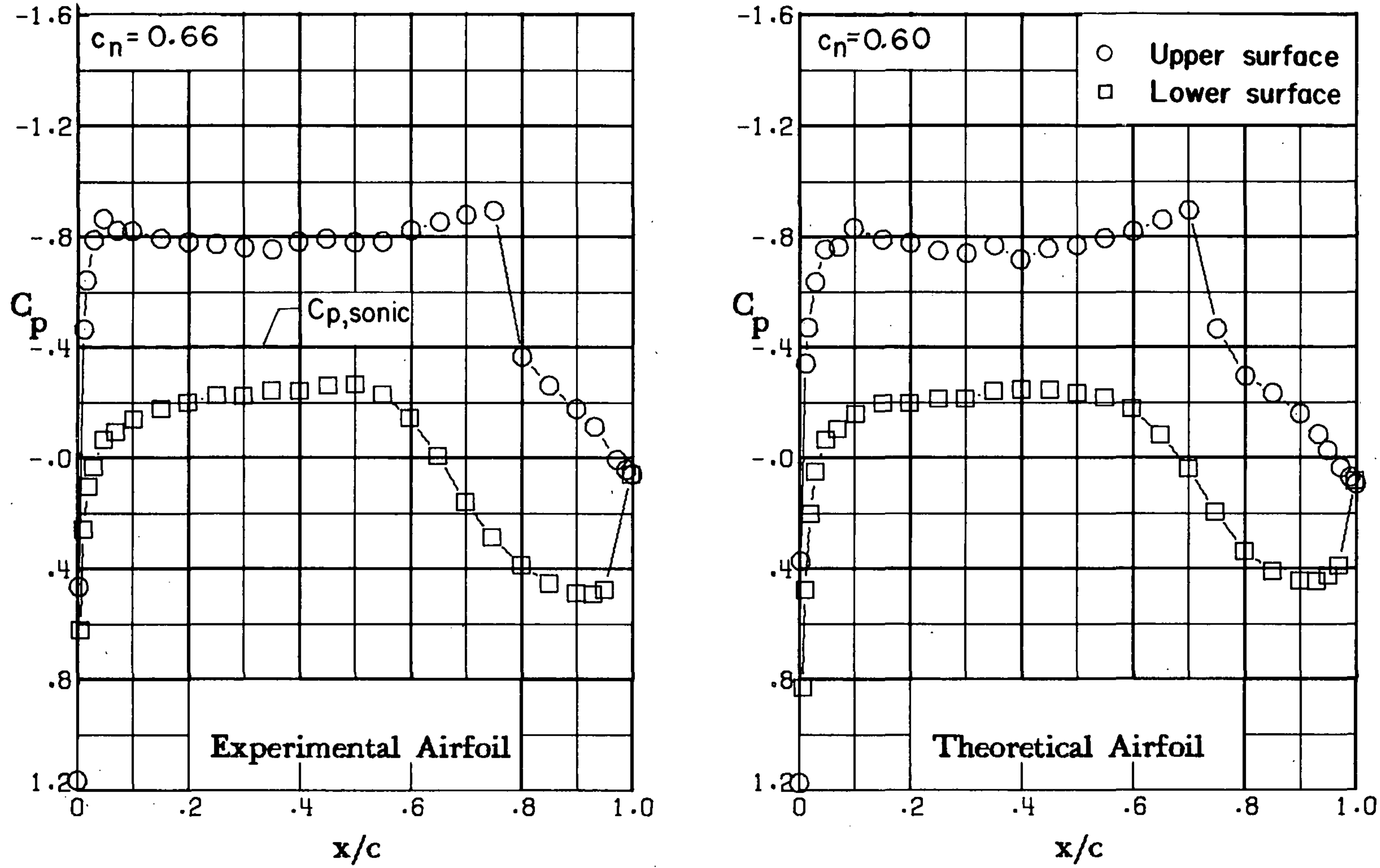
Figure 24.- Continued.

[redacted]



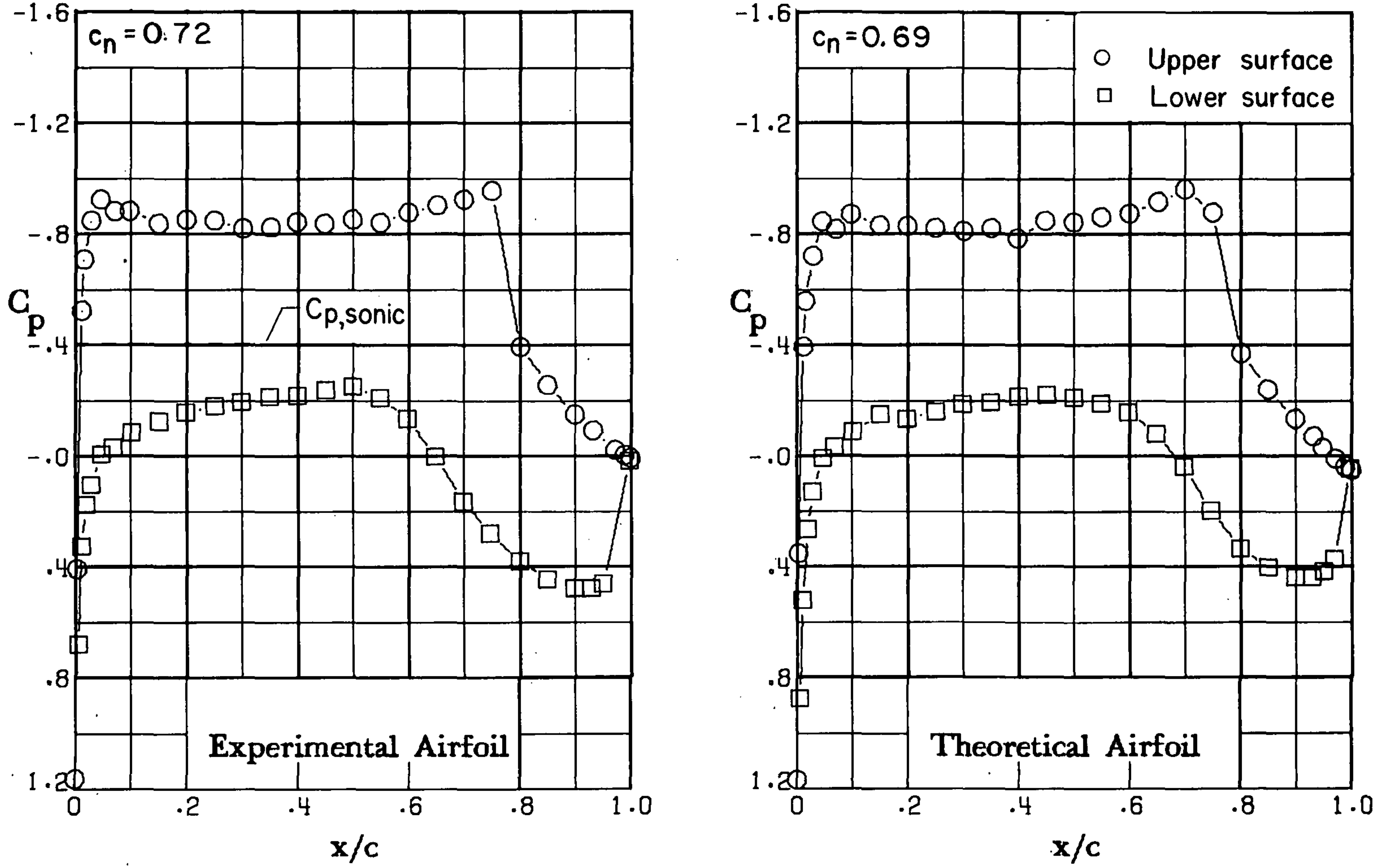
(d) M = 0.81; α = 1.0°.

Figure 24.- Continued.



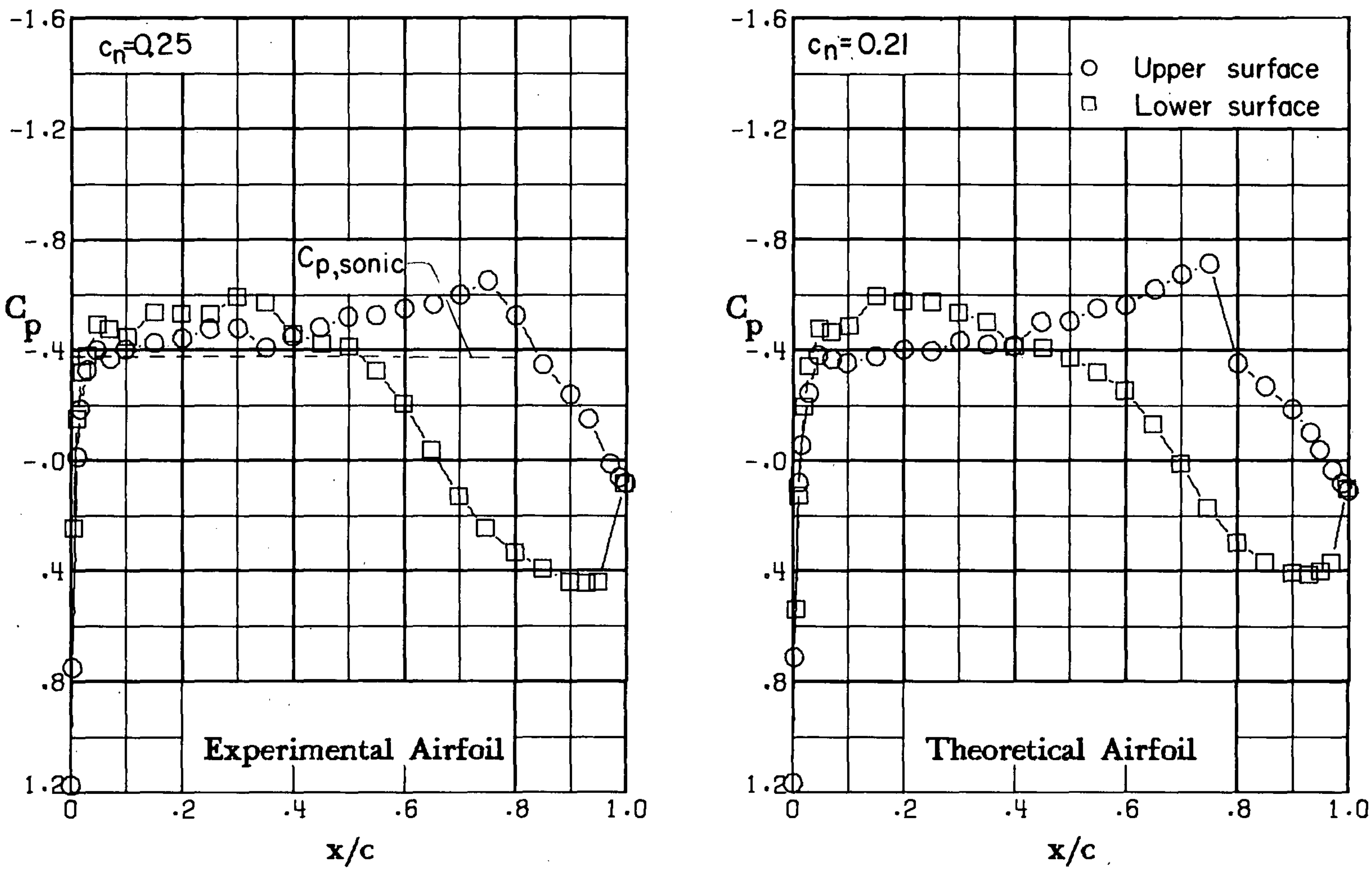
(e) $M = 0.81$; $\alpha = 1.5^{\circ}$.

Figure 24.- Continued.



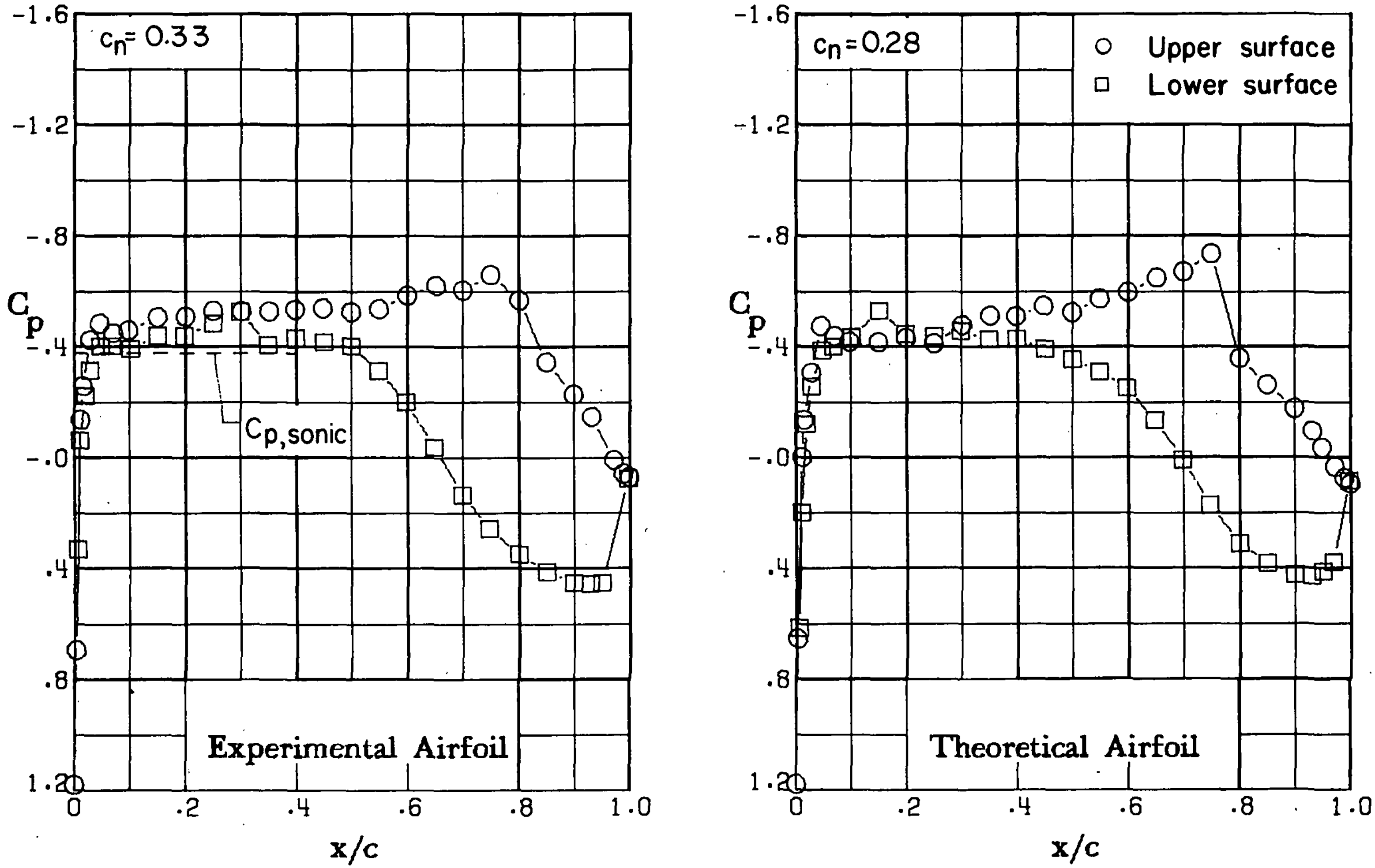
(f) M = 0.81; α = 2.0°.

Figure 24.- Concluded.



(a) M = 0.82; α = -0.5°.

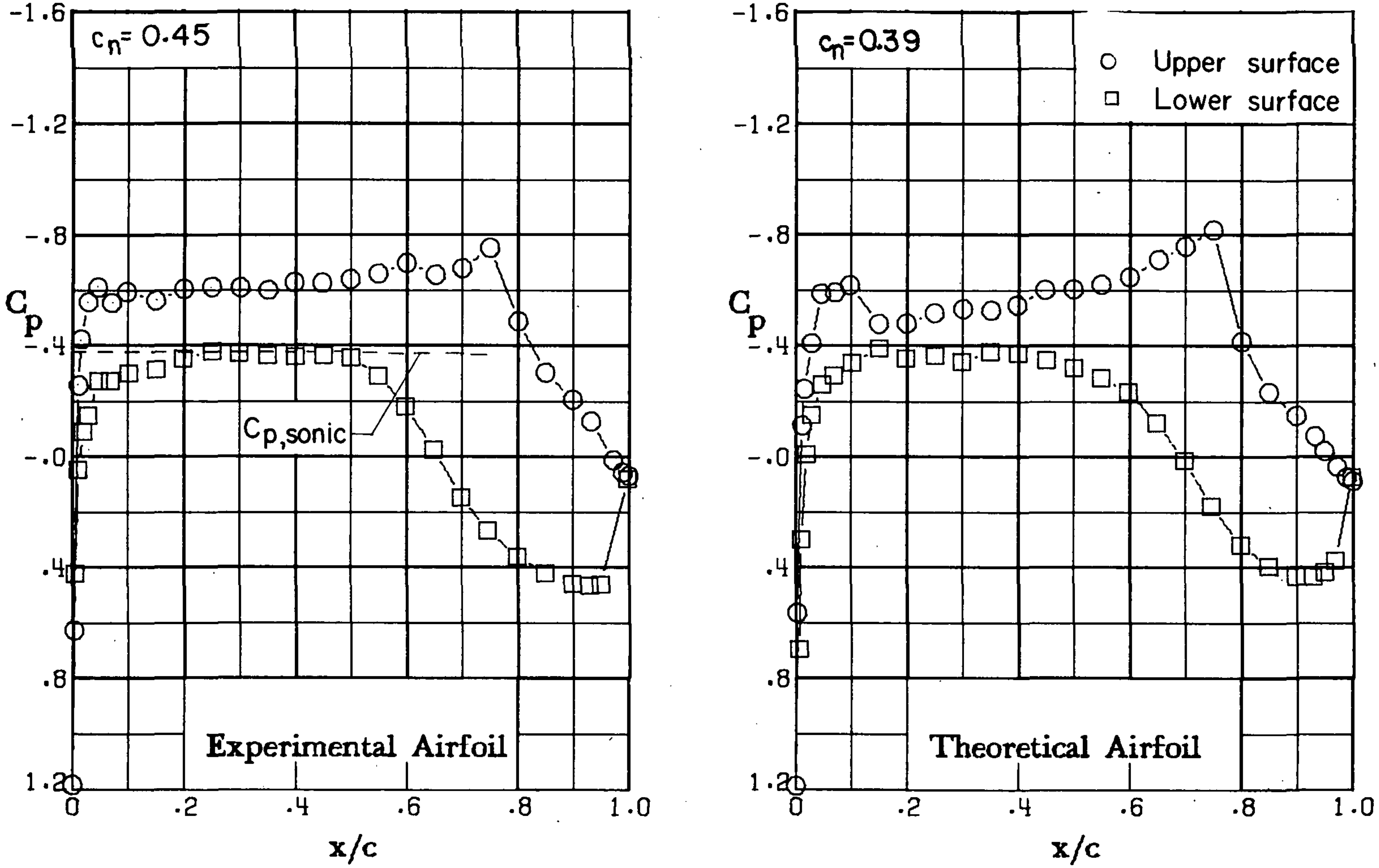
Figure 25.- Chordwise pressure distribution for experimental and theoretical supercritical airfoils.



(b) M = 0.82; α = 0.0°.

Figure 25.- Continued.

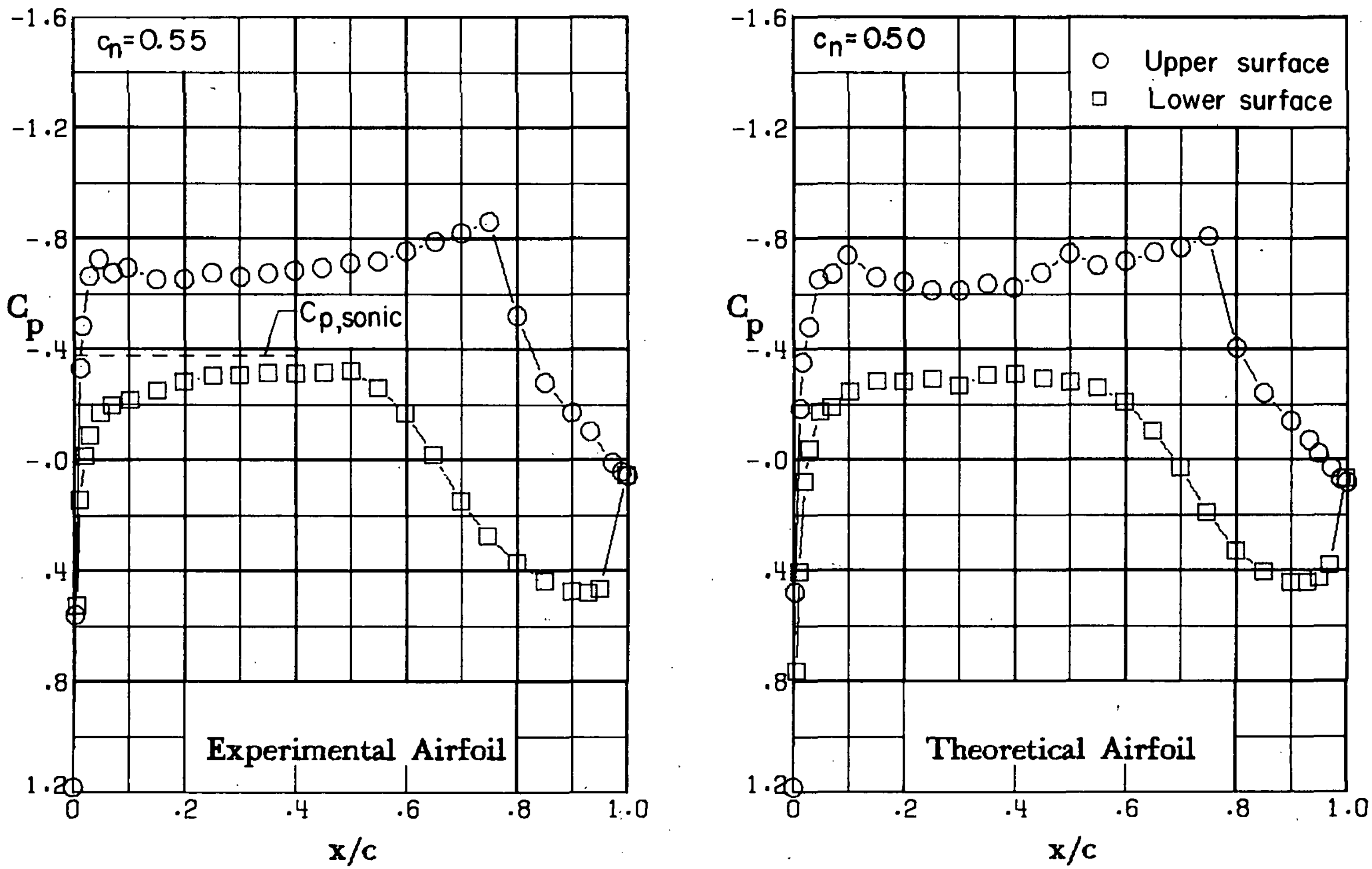
[redacted]



(c) $M = 0.82$; $\alpha = 0.5^{\circ}$.

Figure 25.- Continued.

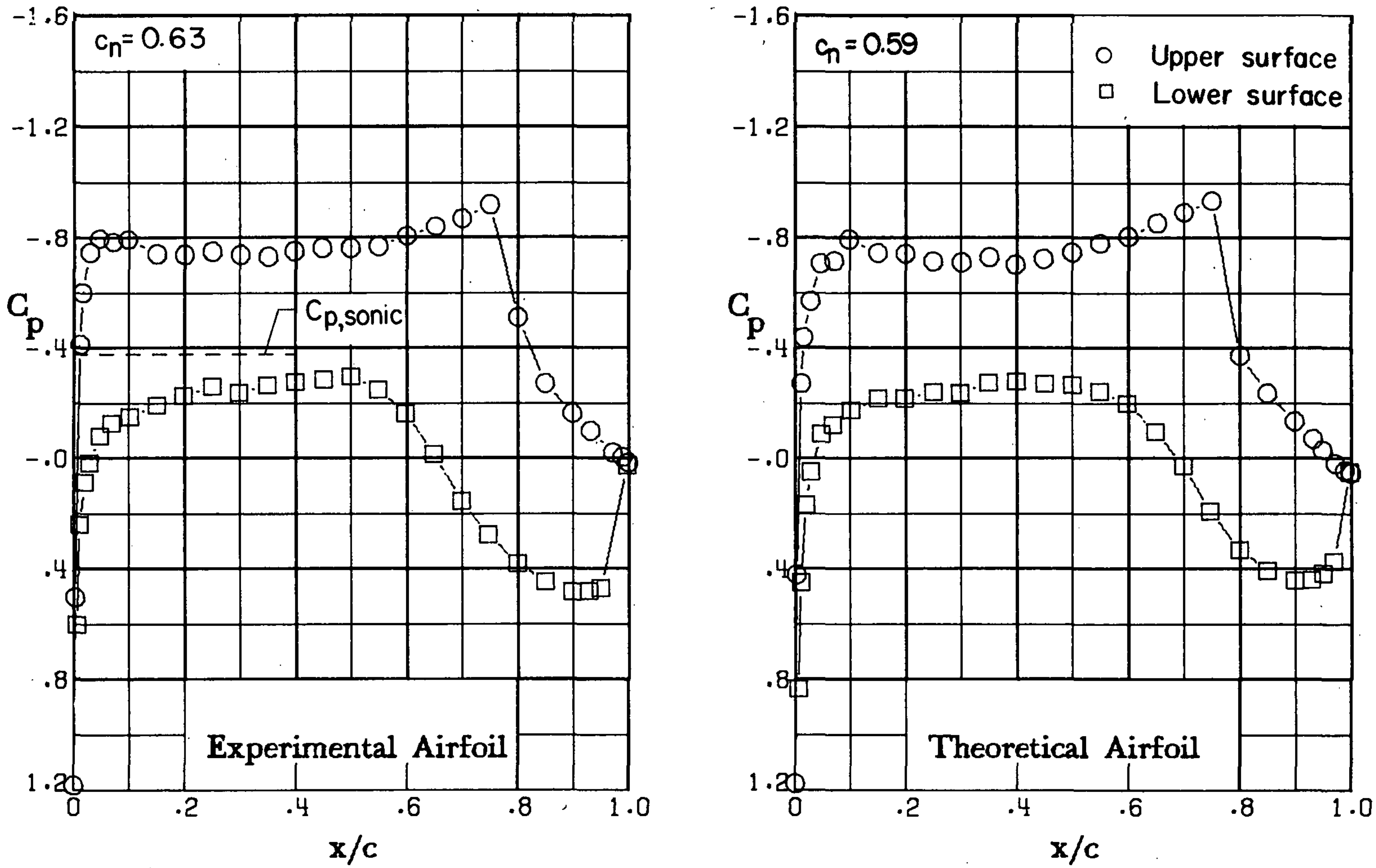
[redacted]



(d) M = 0.82; α = 1.0°.

Figure 25.- Continued.

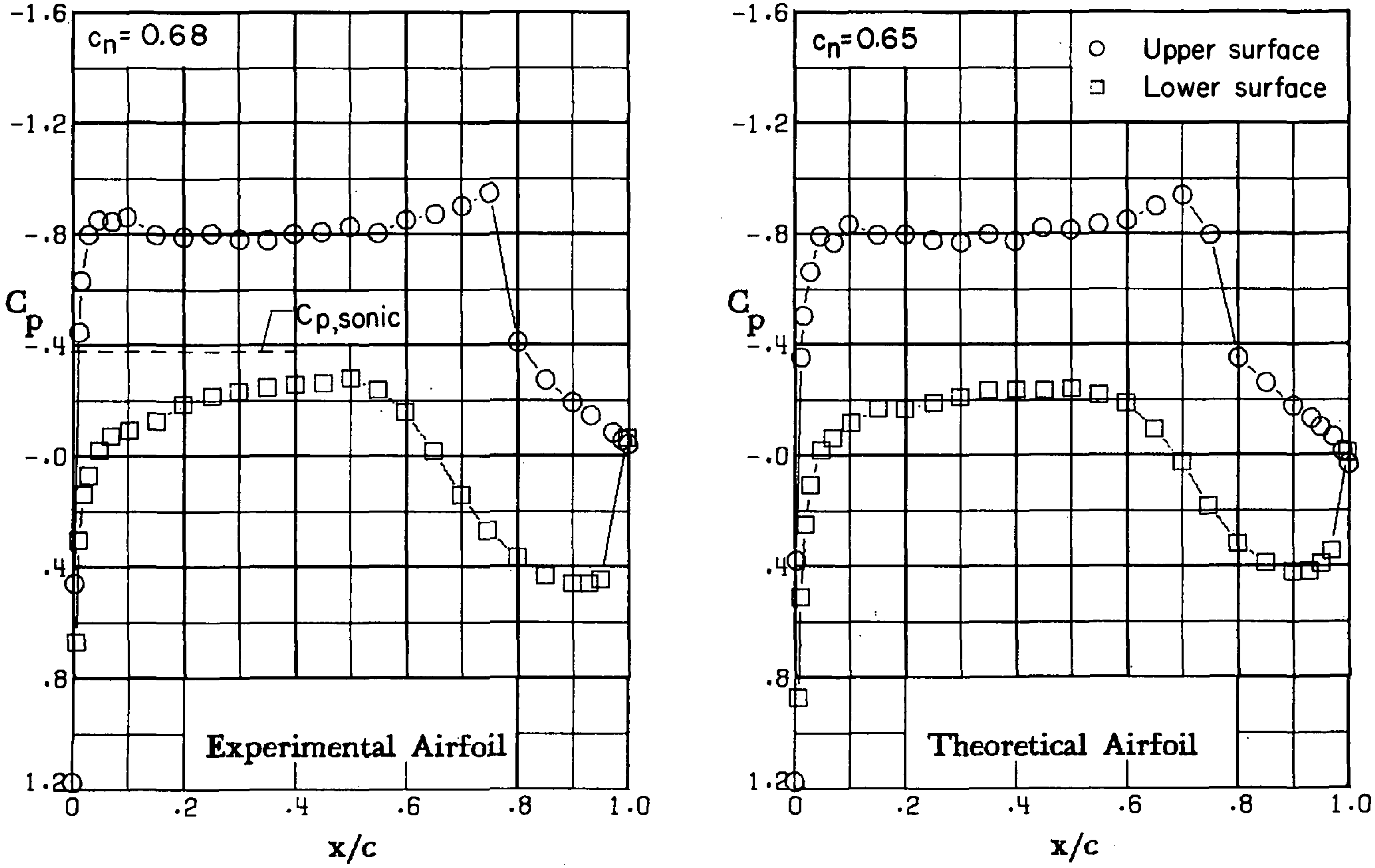
[redacted]



(e) M = 0.82; α = 1.5°.

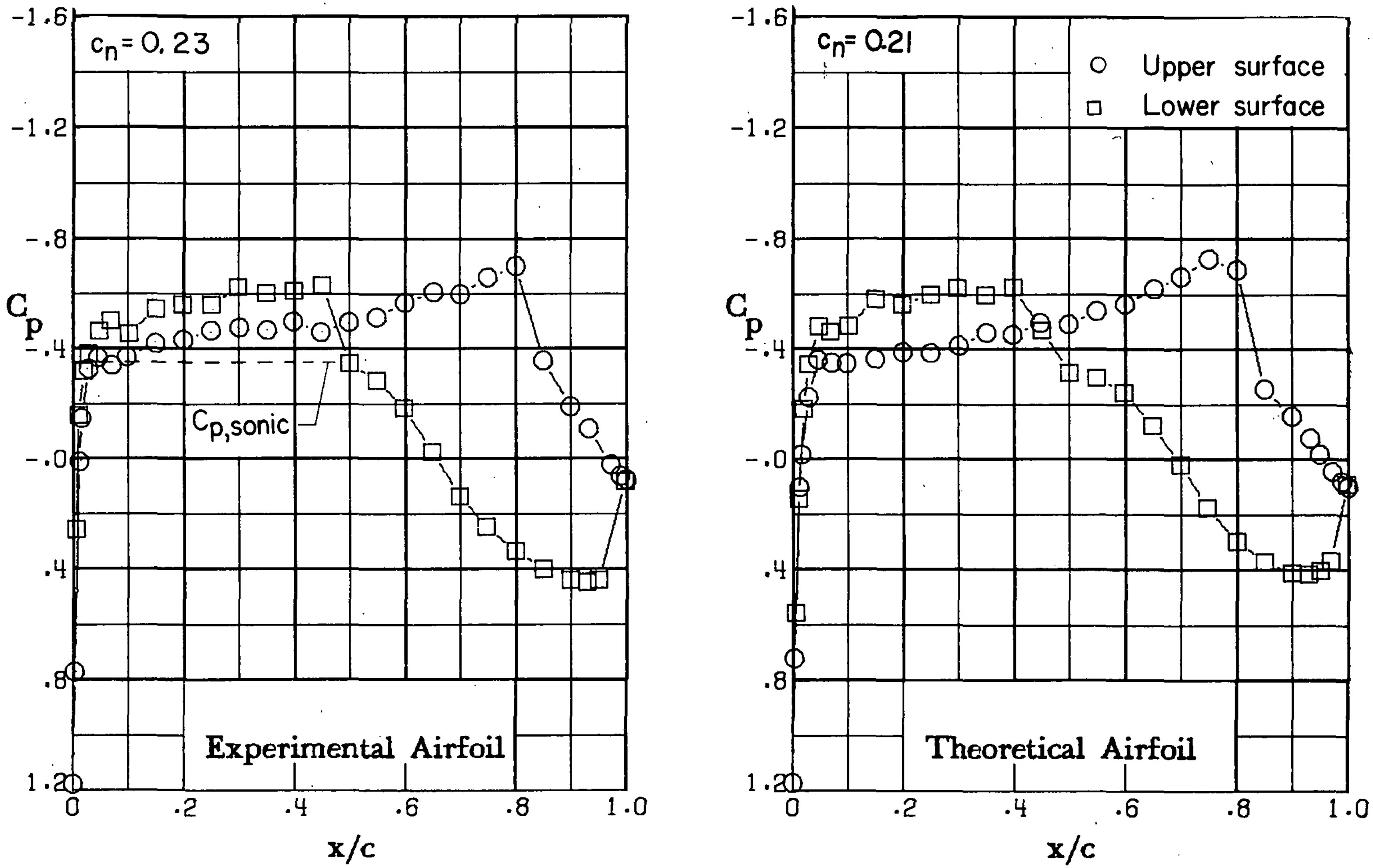
Figure 25.- Continued.

[redacted]



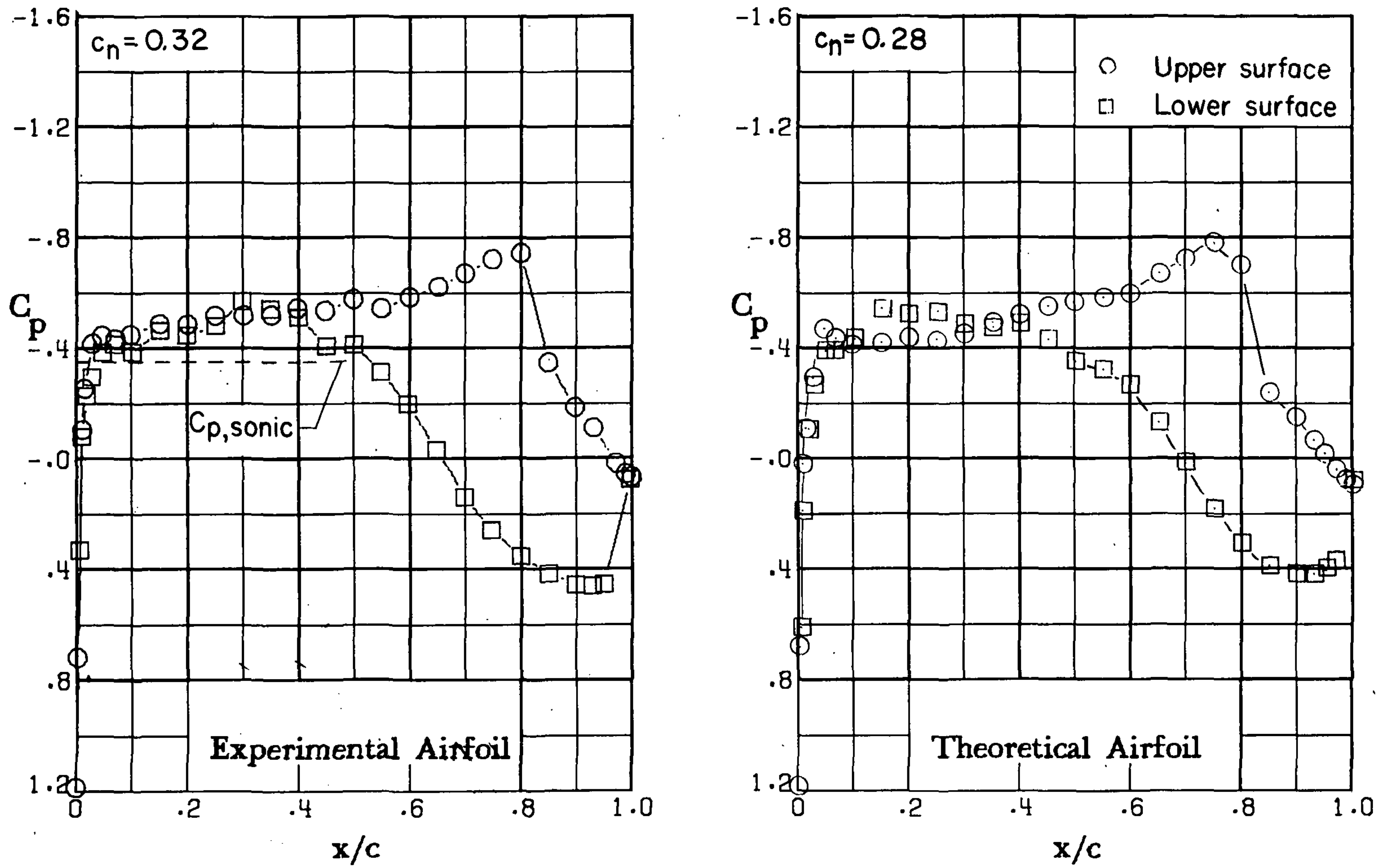
(f) M = 0.82; α = 2.0°.

Figure 25.- Concluded.



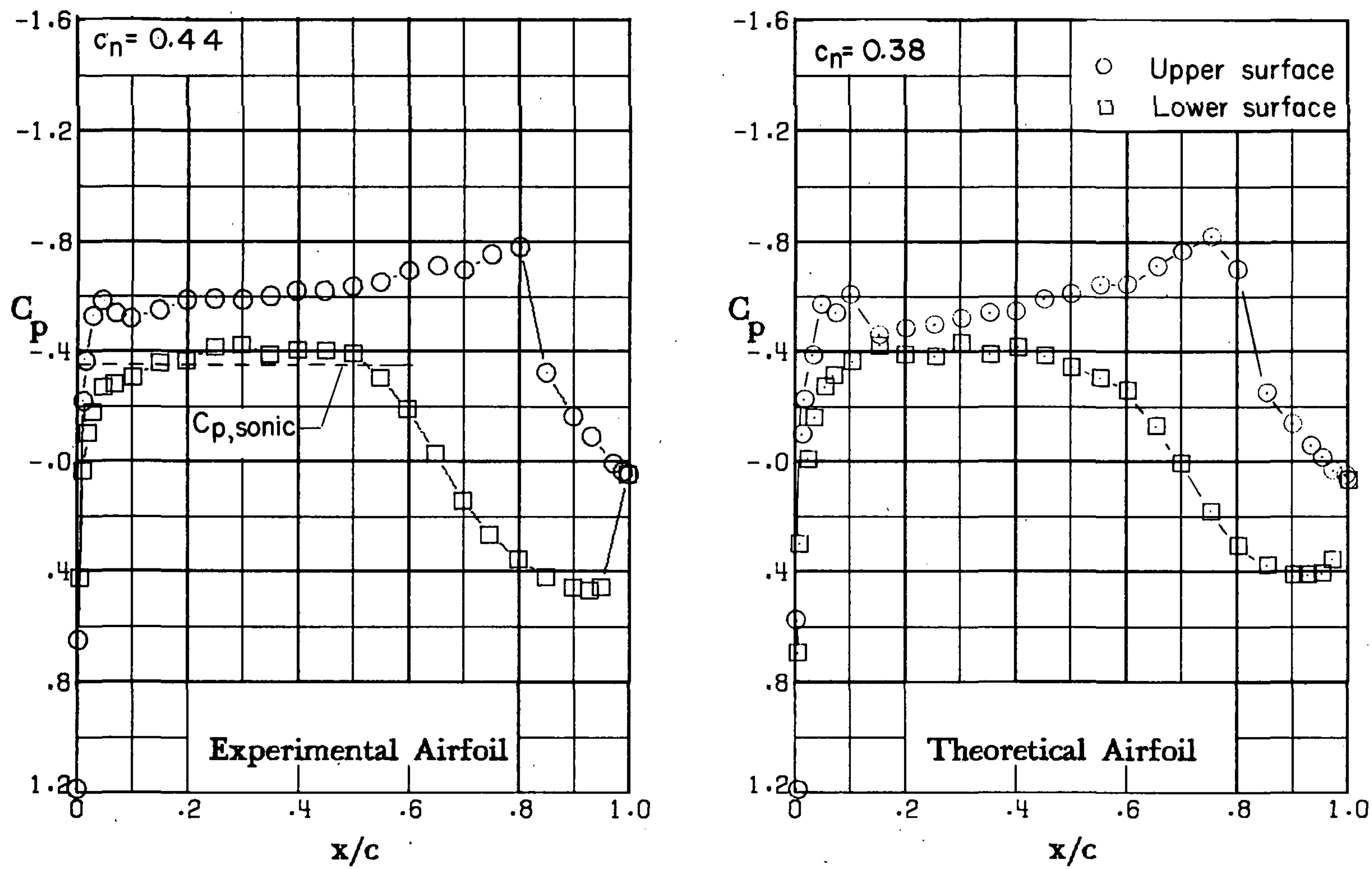
(a) M = 0.83; α = -0.5°.

Figure 26.- Chordwise distribution for experimental and theoretical supercritical airfoils.



(b) $M = 0.83$; $\alpha = 0.0^{o}$.

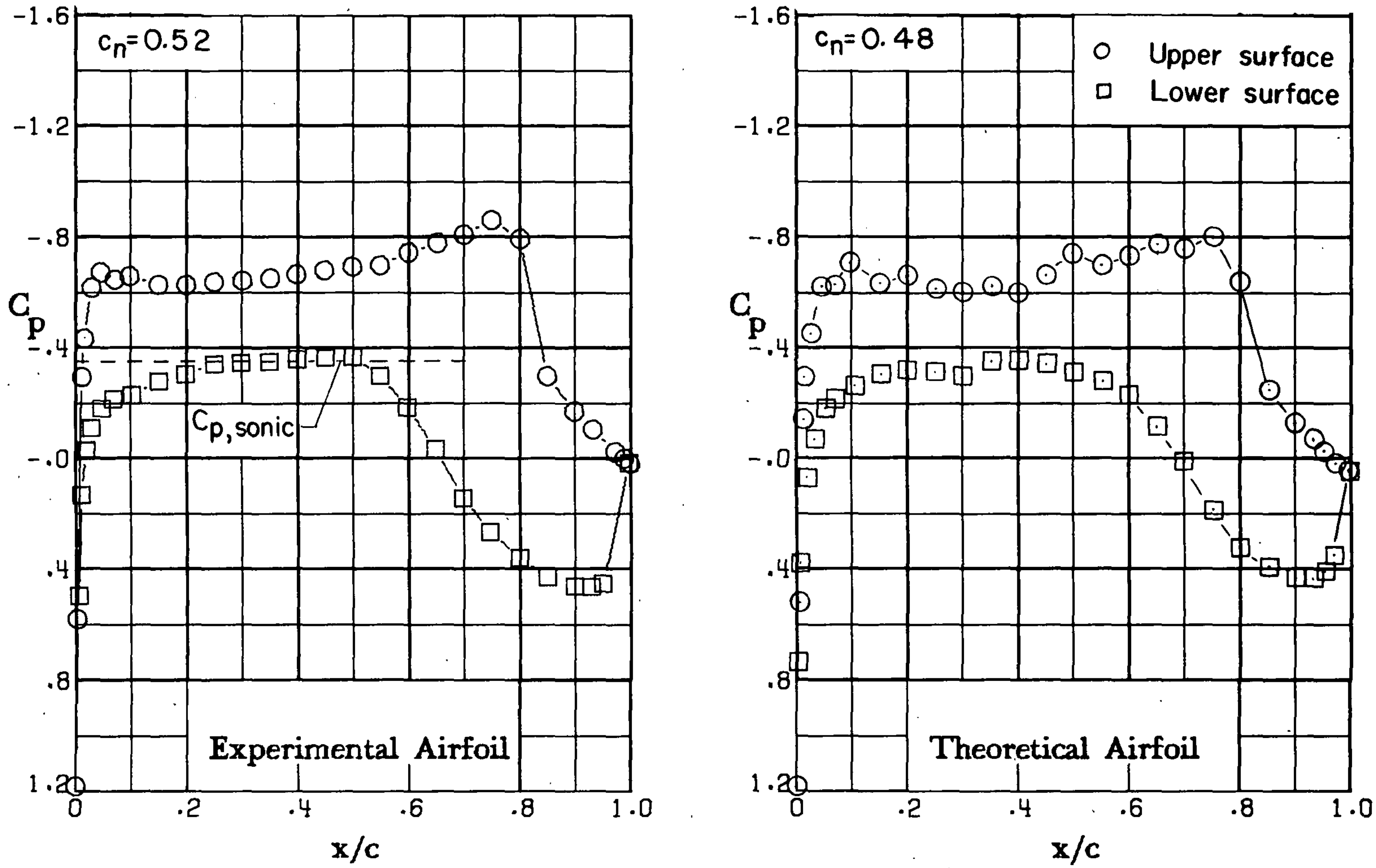
Figure 26.- Continued.



(c) M = 0.83; $\alpha = 0.5^{o}$.

Figure 26.- Continued.

[redacted]



(d) M = 0.83; α = 1.0°.

Figure 26.- Concluded.

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Review for general release September 30, 1982.

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