C 7

N74-12678

[redacted]

NASA TECHNICAL MEMORANDUM

NASA TM X-2978

NASA TM X-2978

[redacted] CLASSIFIED
BY Security Classification Officer, NASA LaRC
SUBJECT TO GENERAL DECLASSIFICATION SCHEDULE OF EXECUTIVE ORDER 11652 AUTOMATICALLY DOWNGRADED AT TWO YEAR INTERVALS AND DECLASSIFIED ON DEC 31 1980

CASE FILE COPY

AERODYNAMIC CHARACTERISTICS OF AN IMPROVED 10-PERCENT-THICK NASA SUPERCRITICAL AIRFOIL

by Charles D. Harris

Langley Research Center
Hampton, Va. 23665

FE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • FEBRUARY 1974

[redacted]

[redacted]

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM X-2978		

4. Title and Subtitle
AERODYNAMIC CHARACTERISTICS OF AN IMPROVED 10-PERCENT-THICK NASA SUPERCRITICAL AIRFOIL (U)

5. Report Date
February 1974

6. Performing Organization Code

7. Author(s)
Charles D. Harris

8. Performing Organization Report No.
L-9275

9. Performing Organization Name and Address
NASA Langley Research Center
Hampton, Va. 23665

10. Work Unit No.
501-06-05-07

11. Contract or Grant No.

12. Sponsoring Agency Name and Address
National Aeronautics and Space Administration
Washington, D.C. 20546

13. Type of Report and Period Covered
Technical Memorandum

14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

This report documents the aerodynamic characteristics of an improved 10-percent-thick supercritical airfoil (airfoil 26a) and compares them with those of earlier supercritical airfoils (airfoils 11 and 12). Integrated section force and moment data, surface-pressure distributions, and typical wake survey profiles are presented.

17. Key Words (Suggested by Author(s))
Supercritical airfoil
Transonic aerodynamics

18. Distribution Statement
[redacted] Available to U.S. Government Agencies and Their Contractors Only

Cat. 01

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
[redacted]	Unclassified	134	

"NATIONAL SECURITY INFORMATION"
Unauthorized Disclosure Subject to Criminal Sanctions.

[redacted] CLASSIFIED
BY Security Classification Officer, NASA LaRC
SUBJECT TO GENERAL DECLASSIFICATION SCHEDULE OF EXECUTIVE ORDER 11652 AUTOMATICALLY DOWNGRADED AT TWO YEAR INTERVALS AND DECLASSIFIED ON DEC 31 1980

[redacted]

[redacted] [redacted]

AERODYNAMIC CHARACTERISTICS OF AN IMPROVED 10-PERCENT-THICK NASA SUPERCRITICAL AIRFOIL*

By Charles D. Harris
Langley Research Center

SUMMARY

Refinements in a 10-percent-thick supercritical airfoil have produced improvements in the overall drag characteristics at normal-force coefficients from about 0.30 to 0.65 compared with earlier supercritical airfoils which were developed for a normal-force coefficient of 0.7. The drag-divergence Mach number of the improved supercritical airfoil (airfoil 26a) varied from approximately 0.82 at a normal-force coefficient of 0.30 to 0.78 at a normal-force coefficient of 0.80 with no drag creep evident.

INTRODUCTION

A concerted effort within the NASA over the past several years, directed toward developing practical two-dimensional airfoils with good transonic behavior while retaining acceptable low-speed characteristics, has focused on a concept referred to as the supercritical airfoil. (See refs. 1 to 7.) This airfoil was conceived to have an extensive region of local supersonic flow over the upper surface terminated by a mainly isentropic recompression near the trailing edge at design conditions; thus, the transonic drag rise is deferred well beyond the critical Mach number (the free-stream Mach number at which the local velocity becomes sonic at some point on the airfoil). A wide range of wind-tunnel and flight investigations (refs. 7 to 13, for example) of several airplane configurations incorporating the supercritical airfoil concept have indicated improvements in performance and maneuver capabilities with marked [redacted]for both military and commercial application of the supercritical airfoil.

Recent research effort has been directed toward two primary objectives: optimization for lift coefficients at which future advanced technology transport configurations are expected [redacted] and reduction of drag creep (gradual buildup of boundary-layer and shock losses preceding drag divergence).

---

*Title, Unclassified.

[redacted]

[redacted]

This report presents the results of this recent effort, documents the aerodynamic characteristics of an improved 10-percent-thick supercritical airfoil (airfoil 26a), and compares these characteristics with those of earlier supercritical airfoils (airfoils 11 and 12).

SYMBOLS

Values are given in both SI and U.S. Customary Units. Measurements and calculations are made in U.S. Customary Units.

C_p	pressure coefficient, $\frac{p_l - p_\infty}{q_\infty}$
$C_{p,sonic}$	pressure coefficient corresponding to local Mach number of 1.0
c	chord of airfoil, 63.5 centimeters (25.0 inches)
c_d	section drag coefficient, $\sum c_d' \frac{\Delta z}{x}$
c_d'	point drag coefficient (ref. 14)
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, $\sum_{l} C_p \frac{\Delta x}{c} - \sum_{u} C_p \frac{\Delta x}{c}$
K	surface curvature, reciprocal of local radius of curvature
M	Mach number
m	surface slope, $\frac{dy}{dx}$
p	static pressure, newtons per meter^2 (pounds per foot^2)
Δp_t	total-pressure loss, newtons per meter^2 (pounds per foot^2)
q	dynamic pressure, newtons per meter^2 (pounds per foot^2)

R Reynolds number based on airfoil chord

x ordinate along airfoil reference line measured from airfoil leading edge, centimeters (inches)

y ordinate normal to airfoil reference line, centimeters (inches)

z vertical distance in wake profile measured from bottom of rake, centimeters (inches)

α angle of attack of airfoil reference line, degrees

Subscripts:

l local point on airfoil

max maximum

∞ undisturbed stream

Abbreviations:

l airfoil lower surface

u airfoil upper surface

APPARATUS AND TECHNIQUES

Descriptions of apparatus and testing techniques contained in this section appear in other reports (refs. 1 to 5, for example), but are repeated here for completeness.

Models

Various features of the supercritical airfoil concept and design philosophy are discussed in references 1 to 7 and are not repeated herein.

Background.- During early phases of the two-dimensional supercritical airfoil development program, emphasis was placed upon developing an airfoil with the highest drag-divergence Mach number attainable at a normal-force coefficient of 0.7. The normal-force coefficient of 0.7 was chosen as the design goal since, when account was taken of the sweep effect, it was representative of lift coefficients at which advanced technology transports utilizing the supercritical airfoil concept were then expected to cruise.

The resultant airfoil, identified as supercritical airfoil 11, with a ratio of maximum thickness to chord of 0.10 and a ratio of trailing-edge thickness to chord of 0.007 had a drag-divergence Mach number of about 0.79 and has been reported in reference 2. This airfoil experienced a drag creep of about 14 counts (c_d increment of 0.0014) between the subcritical Mach number of 0.60 and the final drag-rise conditions. This gradual buildup of drag was largely associated with an intermediate off-design second velocity peak and relatively weak shock waves above the upper surface at these speeds. It was believed that with proper shape refinements, the drag creep could be reduced or eliminated.

Design studies of advanced technology transport configurations suggested that the cruise lift coefficient would be somewhat lower than originally anticipated. Consequently, there was a need for an airfoil optimized for a normal-force coefficient lower than 0.7. The wind-tunnel tests required for airfoil optimization at the lower normal-force coefficient also provided the opportunity to explore the drag-creep problem; thus, drag-creep reduction was included as a goal and an important factor in the wind-tunnel program. The result was an airfoil (identified as airfoil 26a) with a smaller leading-edge radius, reduced curvature over the forward and rear upper surface, reduced aft camber, and minor changes over the lower surface.

Individual systematic modifications explored between airfoils 11 and 26a and included on airfoil 26a are beyond the intended scope of this report and are not discussed.

Wind-tunnel models.- Because data for airfoil 11 were only available at normal-force coefficients near 0.7, there was not an adequate basis of comparison with the improved airfoil 26a at off-design conditions. The first step in the optimization process (airfoil 12) for which data were obtained over a wider range of off-design conditions is used in this report to provide a carryover basis of comparison between the earlier airfoil 11 and the improved airfoil 26a. Physical differences in these three airfoils are shown in figures 1 to 3 and section coordinates are presented in table I. The airfoil number designations were assigned for identification purposes and the airfoils are referred to by these designations hereafter. Although not included in the airfoil sketches of figure 1, a trailing-edge cavity as shown in the photograph of figure 4 and discussed in reference 2 was included on all three airfoils.

The wind-tunnel models, mounted in an inverted position, spanned the width of the tunnel with a span-chord ratio of 3.4. 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 and the profile drag rake are pre-

[redacted]

sented in figure 5 and a photograph of one of the airfoils and the profile drag rake mounted in the tunnel is shown as figure 4(a).

Wind Tunnel

The investigation was conducted in the Langley 8-foot transonic pressure tunnel (ref. 15). 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 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 whereas the slots permit development of the flow field in the vertical direction.

Boundary-Layer Transition

Based on the technique discussed in reference 16, boundary-layer transition was fixed along the 28-percent chord line on the upper and lower surfaces in an attempt to simulate full-scale Reynolds numbers 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-in.) bands of number 90 carborundum grains.

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

[redacted]

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 derived from vertical variations of the total and static pressures measured across the wake with the profile drag rake shown in figure 5. 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. 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 and Corrections

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 14. These point drag coefficients were then summed by numerical integration across the wake, again based on the trapezoidal method.

Corrections for wind-tunnel-wall effects.- The most significant effect of wall interference on the data was a lift-induced angle-of-attack shift which must be subtracted from the measured geometric angle of attack. According to theory (ref. 17), the mean value of the angle-of-attack correction at the midchord, in degrees, is estimated to be approximately three times the section normal-force coefficient. However, based on other experimental data and on estimated differences in angles of attack required to match theoretical and experimental pressure distributions at the same normal-force coefficient, such a cor-

[redacted]

rection is believed to be unrealistically large. Because of this uncertainty, the uncorrected geometric angles of attack are used herein.

The theory of reference 17 also indicates that tunnel-wall-blockage effects would be small; consequently, no corrections have been applied to the data to account for blockage effects.

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) with resultant Reynolds numbers based on the airfoil chord as shown in figure 6. 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.

PRESENTATION OF RESULTS

Section force and moment coefficients of supercritical airfoils 11, 12, and 26a are presented in figure 7 and the drag characteristics are summarized in figures 8 to 11. Although not pertinent to a discussion of the improved airfoil 26a, except as a matter of record in an evolutionary context, comparisons of the section characteristics of airfoils 11 and 12 are included in figures 7 and 8 to provide a source of systematic experimental data for the supercritical airfoil. Chordwise surface-pressure profiles of airfoils 12 and 26a are compared at the same geometric angle of attack in figures 12 to 22 and are presented in tables II and III. (Surface-pressure distributions of airfoil 11 are presented in ref. 2.) Selected wake profiles and surface-pressure profiles are also compared in figures 23 and 24 at various Mach numbers and normal-force coefficients.

The wake profiles in figure 23 are representative of the momentum losses as indicated by stagnation-pressure deficits 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.

DISCUSSION

Neither the drag-rise characteristics summarized in figure 8 nor the surface-pressure distributions (not presented) show any significant differences between airfoils 11 and 12 at near-design normal-force coefficients. Therefore, the remainder of this discussion will be confined to comparisons of the aerodynamic characteristics of airfoils 12 and 26a.

[redacted]

Figure 7 shows an increase of almost 1° in angle of attack to achieve the same normal-force coefficient with airfoil 26a as with airfoil 12 because of the reduced aft camber. There was also a substantial reduction in nose-down pitching moments associated with this reduced aft camber.

Drag-divergence Mach number was significantly improved (figs. 9 and 10) and varied from approximately 0.82 at $c_n = 0.3$ to approximately 0.78 at $c_n = 0.8$ for airfoil 26a. For the comparison of figure 10, drag-divergence Mach number was defined as the point where the slope of the curve for section drag coefficient as a function of Mach number equals 0.1. The improvements in drag-divergence Mach number for airfoil 26a were associated with the elimination of a region of overexpansion on the upper surface of airfoil 12 near the 80-percent chord station. This region of overexpansion on airfoil 12 developed into a second supersonic velocity peak and terminated in a second shock system at the higher Mach numbers which accounts for the lower drag-divergence Mach number. (See, for example, fig. 19(c).) This region of overexpansion and subsequent second supersonic velocity peak is no longer evident on airfoil 26a. These improved drag-rise characteristics of airfoil 26a may best be illustrated by the wake profiles in figures 23(c) to 23(f). It may be seen that the delays in drag-divergence Mach number for airfoil 26a at the lower normal-force coefficients were due to noticeable reductions in both viscous and shock losses.

Airfoil 26a also exhibited (figs. 9 and 11) a lower subcritical drag level than airfoil 12 and no drag creep was evident up to normal-force coefficients somewhat greater than 0.6. In fact, for these lower normal-force coefficients, figure 9 shows a decrease in drag with increasing Mach number due to the variation of Reynolds number. (See fig. 6.) If the Reynolds number influence were taken into account, the drag curves of figure 9 would be practically flat up to the drag-divergence Mach number. The lower subcritical drag levels may be explained by referring to the surface-pressure distributions. Compare, for example, the surface-pressure distributions of the two airfoils at the same normal-force coefficients for Mach numbers of 0.50 and 0.60 in figures 24(a) and 24(b). Airfoil 26a, although starting with a higher leading-edge suction peak than airfoil 12, developed a gradual recompression from the leading-edge suction peak to the trailing edge that resulted in reduced adverse pressure gradients at the trailing edge where boundary-layer development is most sensitive.

At normal-force coefficients greater than about 0.7, the drag-rise characteristics (fig. 10) of airfoil 26a were not as good as those of airfoil 12. This result was not unexpected since the aft camber of airfoil 12 was designed for higher normal-force coefficients. At Mach numbers below the drag rise, the drag of airfoil 26a was generally greater than airfoil 12 (fig. 7) because of the reduced aft camber. Due to the larger angles of attack required with airfoil 26a to obtain the same lift as airfoil 12, supervelocities were greater over the forward upper surface region of airfoil 26a and resulted in stronger shock strengths. (See figs. 24(c) and 24(d).) The stronger shock strengths are also illustrated by the wake-profile comparisons at these conditions in figures 23(a) and 23(b).

CONCLUDING REMARKS

Refinements in a 10-percent-thick supercritical airfoil have produced improvements in the overall drag characteristics at normal-force coefficients from about 0.30 to 0.65 compared with earlier supercritical airfoils which were developed for a normal-force coefficient of 0.7. The drag-divergence Mach number of the improved supercritical airfoil (airfoil 26a) varied from approximately 0.82 at a normal-force coefficient of 0.30 to 0.78 at a normal-force coefficient of 0.80 with no drag creep evident.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., November 28, 1973.

[redacted]

REFERENCES

1. Whitcomb, Richard T.; and Clark, Larry R.: An Airfoil Shape for Efficient Flight at Supercritical Mach Numbers. NASA TM X-1109, 1965.

2. Harris, Charles D.: Wind-Tunnel Investigation of Effects of Trailing-Edge Geometry on a NASA Supercritical Airfoil Section. NASA TM X-2336, 1971.

3. Harris, Charles D.; and Blackwell, James A., Jr.: Wind-Tunnel Investigation of Effects of Rear Upper Surface Modification on an NASA Supercritical Airfoil. NASA TM X-2454, 1972.

4. Harris, Charles D.: Aerodynamic Characteristics of Two NASA Supercritical Airfoils With Different Maximum Thicknesses. NASA TM X-2532, 1972.

5. Harris, Charles D.: Aerodynamic Characteristics of Two 10-Percent-Thick NASA Supercritical Airfoils With Different Upper Surface Curvature Distributions. NASA TM X-2977, 1974.

6. McGhee, Robert J.; and Bingham, Gene J.: Low-Speed Aerodynamic Characteristics of a 17-Percent-Thick Supercritical Airfoil Section, Including a Comparison Between Wind-Tunnel and Flight Data. NASA TM X-2571, 1972.

7. Anon.: Supercritical Wing Technology – A Progress Report on Flight Evaluations. NASA SP-301, 1972.

8. Bartlett, Dennis W.; and Re, Richard J.: Wind-Tunnel Investigation of Basic Aerodynamic Characteristics of a Supercritical-Wing Research Airplane Configuration. NASA TM X-2470, 1972.

9. Palmer, W. E.; Elliott, D. W.; and White, J. E.: Flight and Wind Tunnel Evaluation of a 17% Thick Supercritical Airfoil on a T-2C Airplane. NR71H-150 (Contract N00019-70-C-0474), North American Rockwell Corp., July 31, 1971.
   Vol. I – Basic Report. (Available from DDC as AD 517 436L.)
   Vol. II – Flight Measured Wing Wake Profiles and Surface Pressures. (Available from DDC as AD 517 437L.)

10. Hallissy, James B.: Longitudinal Aerodynamic Characteristics of a Carrier-Based Variable-Wing-Sweep Fighter Airplane With Partial Incorporation of the NASA Supercritical Airfoil. NASA TM X-2745, 1973.

11. Langhans, Richard A.; and Flechner, Stuart G.: Wind-Tunnel Investigation at Mach Numbers From 0.25 to 1.01 of a Transport Configuration Designed To Cruise at Near-Sonic Speeds. NASA TM X-2622, 1972.

[redacted]

12. Ferris, James C.: Static Aerodynamic Characteristics of a Model With a 17-Percent-Thick Supercritical Wing. NASA TN X-2551, 1972.

13. Ayers, Theodore G.: A Wind-Tunnel Investigation of the Application of the NASA Supercritical Airfoil to a Variable-Wing-Sweep Fighter Airplane. NASA TM X-2759, 1973.

14. Baals, Donald D.; and Mourhess, Mary J.: Numerical Evaluation of the Wake-Survey Equations for Subsonic Flow Including the Effect of Energy Addition. NACA WR L-5, 1945. (Formerly NACA ARR L5H27.)

15. Schaefer, William T., Jr.: Characteristics of Major Active Wind Tunnels at the Langley Research Center. NASA TM X-1130, 1965.

16. Blackwell, James A., Jr.: Preliminary Study of Effects of Reynolds Number and Boundary-Layer Transition Location on Shock-Induced Separation. NASA TN D-5003, 1969.

17. Davis, Don D., Jr.; and Moore, Dewey: Analytical Study of Blockage- and Lift-Interference Corrections for Slotted Tunnels Obtained by the Substitution of an Equivalent Homogeneous Boundary for the Discrete Slots. NACA RM L53E07b, 1953.

[redacted]

TABLE I.- SECTION COORDINATES

[c = 63.5 cm (25 in.); airfoils 11 and 12 have a leading-edge radius of 0.0212c; airfoil 26a has an upper surface leading-edge radius of 0.0174c and a lower surface leading-edge radius of 0.0189c]

x/c	$(y/c)_u$	$(y/c)_l$	$(y/c)_u$	$(y/c)_l$	$(y/c)_u$	$(y/c)_l$	x/c	$(y/c)_u$	$(y/c)_l$	$(y/c)_u$	$(y/c)_l$	$(y/c)_u$	$(y/c)_l$
	Airfoil 11		Airfoil 12		Airfoil 26a			Airfoil 11		Airfoil 12		Airfoil 26a	
0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.50	0.0493	-0.0469	0.0493	-0.0468	0.0493	-0.0463
.005	.0137	-.0137	.0137	-.0137	.0122	-.0128	.51	.0492	-.0465	.0492	-.0463	.0491	-.0457
.01	.0181	-.0182	.0181	-.0182	.0163	-.0169	.52	.0490	-.0460	.0490	-.0458	.0489	-.0450
.02	.0232	-.0237	.0232	-.0237	.0212	-.0217	.53	.0489	-.0455	.0489	-.0452	.0487	-.0443
.03	.0267	-.0274	.0267	-.0274	.0244	-.0250	.54	.0487	-.0449	.0487	-.0446	.0485	-.0435
.04	.0294	-.0304	.0294	-.0304	.0269	-.0276	.55	.0485	-.0442	.0485	-.0438	.0482	-.0427
.05	.0316	-.0328	.0316	-.0328	.0290	-.0298	.56	.0483	-.0435	.0482	-.0430	.0479	-.0418
.06	.0335	-.0348	.0335	-.0348	.0308	-.0317	.57	.0480	-.0427	.0480	-.0421	.0476	-.0408
.07	.0351	-.0365	.0351	-.0365	.0324	-.0334	.58	.0478	-.0418	.0478	-.0411	.0473	-.0397
.08	.0366	-.0381	.0366	-.0381	.0339	-.0349	.59	.0475	-.0407	.0475	-.0400	.0470	-.0386
.09	.0378	-.0394	.0378	-.0394	.0352	-.0363	.60	.0472	-.0396	.0472	-.0388	.0466	-.0374
.10	.0390	-.0406	.0390	-.0406	.0364	-.0376	.61	.0469	-.0383	.0469	-.0375	.0462	-.0361
.11	.0401	-.0417	.0401	-.0417	.0375	-.0388	.62	.0466	-.0369	.0465	-.0360	.0458	-.0347
.12	.0410	-.0427	.0410	-.0427	.0385	-.0399	.63	.0463	-.0354	.0462	-.0343	.0454	-.0332
.13	.0419	-.0435	.0419	-.0435	.0395	-.0409	.64	.0459	-.0336	.0458	-.0325	.0450	-.0316
.14	.0427	-.0443	.0427	-.0443	.0404	-.0418	.65	.0456	-.0316	.0454	-.0305	.0445	-.0299
.15	.0434	-.0450	.0434	-.0450	.0412	-.0426	.66	.0452	-.0295	.0450	-.0282	.0440	-.0282
.16	.0440	-.0457	.0440	-.0457	.0420	-.0434	.67	.0447	-.0271	.0445	-.0258	.0435	-.0264
.17	.0447	-.0462	.0447	-.0462	.0427	-.0441	.68	.0443	-.0247	.0440	-.0234	.0430	-.0246
.18	.0452	-.0468	.0452	-.0468	.0434	-.0448	.69	.0438	-.0224	.0435	-.0210	.0424	-.0227
.19	.0457	-.0472	.0457	-.0472	.0440	-.0454	.70	.0433	-.0201	.0430	-.0188	.0418	-.0208
.20	.0462	-.0476	.0462	-.0476	.0446	-.0460	.71	.0427	-.0179	.0424	-.0167	.0412	-.0189
.21	.0466	-.0480	.0466	-.0480	.0452	-.0465	.72	.0421	-.0158	.0418	-.0146	.0406	-.0170
.22	.0470	-.0484	.0470	-.0484	.0457	-.0470	.73	.0415	-.0137	.0411	-.0126	.0399	-.0151
.23	.0474	-.0487	.0474	-.0487	.0462	-.0474	.74	.0409	-.0118	.0404	-.0107	.0392	-.0132
.24	.0477	-.0489	.0477	-.0489	.0466	-.0478	.75	.0402	-.0100	.0397	-.0090	.0385	-.0114
.25	.0480	-.0492	.0480	-.0492	.0470	-.0482	.76	.0394	-.0082	.0389	-.0073	.0377	-.0096
.26	.0483	-.0494	.0483	-.0494	.0474	-.0485	.77	.0386	-.0066	.0380	-.0057	.0369	-.0078
.27	.0486	-.0495	.0486	-.0495	.0477	-.0488	.78	.0378	-.0050	.0371	-.0042	.0361	-.0061
.28	.0488	-.0497	.0488	-.0497	.0480	-.0491	.79	.0369	-.0036	.0361	-.0028	.0352	-.0044
.29	.0490	-.0498	.0490	-.0498	.0483	-.0493	.80	.0359	-.0023	.0351	-.0015	.0343	-.0028
.30	.0492	-.0499	.0492	-.0499	.0486	-.0495	.81	.0348	-.0010	.0340	-.0003	.0333	-.0013
.31	.0493	-.0499	.0493	-.0499	.0488	-.0497	.82	.0337	.0001	.0329	+.0007	.0323	+.0001
.32	.0495	-.0500	.0495	-.0500	.0490	-.0498	.83	.0326	.0011	.0316	.0017	.0312	.0014
.33	.0496	-.0500	.0496	-.0500	.0492	-.0499	.84	.0313	.0019	.0303	.0025	.0301	.0026
.34	.0497	-.0500	.0497	-.0500	.0494	-.0500	.85	.0299	.0027	.0289	.0031	.0289	.0036
.35	.0498	-.0500	.0498	-.0500	.0496	-.0500	.86	.0285	.0032	.0275	.0036	.0277	.0045
.36	.0499	-.0499	.0499	-.0499	.0497	-.0500	.87	.0270	.0037	.0259	.0040	.0264	.0052
.37	.0499	-.0499	.0499	-.0499	.0498	-.0500	.88	.0253	.0039	.0242	.0042	.0250	.0057
.38	.0500	-.0498	.0500	-.0498	.0499	-.0499	.89	.0235	.0040	.0224	.0042	.0235	.0060
.39	.0500	-.0497	.0500	-.0497	.0500	-.0498	.90	.0217	.0039	.0206	.0040	.0219	.0061
.40	.0500	-.0495	.0500	-.0495	.0500	-.0497	.91	.0196	.0036	.0186	.0036	.0202	.0061
.41	.0500	-.0494	.0500	-.0494	.0500	-.0495	.92	.0175	.0030	.0164	.0030	.0184	.0059
.42	.0500	-.0492	.0500	-.0492	.0500	-.0493	.93	.0152	.0022	.0142	.0021	.0165	.0054
.43	.0499	-.0490	.0499	-.0490	.0500	-.0491	.94	.0128	+.0011	.0118	.0010	.0145	.0046
.44	.0499	-.0488	.0499	-.0488	.0500	-.0488	.95	.0102	-.0002	.0093	-.0005	.0124	.0035
.45	.0498	-.0486	.0498	-.0486	.0499	-.0485	.96	.0074	-.0019	.0066	-.0022	.0102	.0021
.46	.0498	-.0483	.0498	-.0483	.0498	-.0482	.97	.0044	-.0040	.0038	-.0043	.0079	+.0004
.47	.0497	-.0480	.0497	-.0480	.0497	-.0478	.98	+.0012	-.0065	.0008	-.0067	.0055	-.0016
.48	.0496	-.0477	.0496	-.0476	.0496	-.0474	.99	-.0021	-.0094	-.0024	-.0095	.0029	-.0039
.49	.0495	-.0473	.0495	-.0472	.0495	-.0469	1.00	------	-.0127	------	-.0127	------	-.0066

[redacted]

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12

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

	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	1.066	1.096	1.124	1.140	1.146	1.156	1.163	1.162	1.166	1.171		0.000
.003	.363	.464	.580	.627	.681	.693	.710	.727	.744	.755		.003
.012	-.510	-.431	-.300	-.239	-.155	-.124	-.116	-.062	-.028	-.011		.012
.019	-.658	-.606	-.539	-.480	-.421	-.399	-.342	-.329	-.289	-.275		.019
.028	-.653	-.616	-.597	-.557	-.525	-.508	-.477	-.448	-.416	-.387		.028
.047	-.571	-.557	-.556	-.550	-.514	-.489	-.485	-.455	-.444	-.427		.047
.072	-.543	-.546	-.588	-.598	-.600	-.581	-.568	-.556	-.516	-.502		.072
.098	-.478	-.499	-.496	-.533	-.536	-.543	-.553	-.578	-.551	-.548		.098
.150	-.394	-.410	-.465	-.494	-.499	-.513	-.522	-.532	-.525	-.523		.150
.200	-.378	-.392	-.412	-.445	-.466	-.482	-.482	-.488	-.481	-.491		.200
.250	-.363	-.377	-.391	-.409	-.417	-.442	-.470	-.476	-.493	-.488		.250
.301	-.343	-.357	-.383	-.401	-.391	-.402	-.391	-.376	-.375	-.452		.301
.351	-.333	-.344	-.374	-.391	-.391	-.395	-.399	-.392	-.387	-.361		.351
.398	-.322	-.339	-.366	-.377	-.388	-.393	-.396	-.392	-.391	-.389		.398
.448	-.335	-.343	-.372	-.391	-.395	-.402	-.409	-.401	-.402	-.393		.448
.499	-.341	-.356	-.385	-.412	-.411	-.430	-.432	-.428	-.427	-.428		.499
.549	-.351	-.357	-.403	-.423	-.437	-.452	-.450	-.447	-.444	-.439		.549
.600	-.385	-.397	-.439	-.466	-.486	-.500	-.502	-.503	-.499	-.488		.600
.652	-.379	-.394	-.444	-.473	-.495	-.516	-.529	-.536	-.537	-.532		.652
.700	-.400	-.424	-.472	-.514	-.543	-.576	-.572	-.589	-.584	-.579		.700
.750	-.417	-.443	-.499	-.546	-.578	-.639	-.654	-.656	-.657	-.647		.750
.801	-.420	-.439	-.502	-.534	-.574	-.656	-.739	-.752	-.755	-.749		.801
.850	-.417	-.428	-.465	-.488	-.489	-.483	-.481	-.561	-.682	-.699		.850
.899	-.345	-.354	-.361	-.355	-.338	-.315	-.276	-.254	-.228	-.225		.899
.932	-.270	-.259	-.247	-.232	-.212	-.183	-.156	-.138	-.123	-.140		.932
.949	-.208	-.190	-.177	-.149	-.132	-.108	-.089	-.076	-.078	-.107		.949
.972	-.107	-.081	-.060	-.047	-.031	-.021	-.017	-.020	-.032	-.046		.972
.981	-.069	-.047	-.019	-.011	-.001	.002	.010	.001	-.019	-.061		.981
1.000	-.006	.015	.020	.027	.034	.031	.028	.020	.001	-.024		1.000
					LOWER SURFACE							
.006	.265	.283	.286	.277	.262	.289	.301	.282	.280	.280		.006
.011	.004	-.018	-.033	-.039	-.052	-.044	-.042	-.034	-.034	-.023		.011
.020	-.180	-.211	-.250	-.239	-.265	-.263	-.247	-.260	-.269	-.254		.020
.029	-.261	-.306	-.355	-.382	-.405	-.400	-.426	-.411	-.424	-.407		.029
.048	-.278	-.343	-.403	-.439	-.457	-.467	-.480	-.491	-.497	-.496		.048
.069	-.280	-.337	-.408	-.455	-.501	-.536	-.539	-.550	-.546	-.540		.069
.102	-.290	-.329	-.390	-.454	-.485	-.540	-.559	-.619	-.640	-.645		.102
.151	-.278	-.300	-.374	-.408	-.441	-.494	-.501	-.564	-.662	-.677		.151
.199	-.237	-.259	-.304	-.340	-.355	-.386	-.398	-.410	-.583	-.669		.199
.251	-.178	-.216	-.248	-.279	-.295	-.323	-.335	-.340	-.294	-.583		.251
.299	-.177	-.205	-.251	-.267	-.284	-.311	-.329	-.337	-.346	-.266		.299
.350	-.158	-.184	-.220	-.243	-.255	-.273	-.287	-.303	-.313	-.296		.350
.400	-.153	-.176	-.215	-.234	-.246	-.255	-.268	-.287	-.299	-.296		.400
.451	-.158	-.181	-.216	-.239	-.244	-.257	-.268	-.283	-.297	-.306		.451
.500	-.180	-.203	-.234	-.250	-.259	-.279	-.291	-.298	-.320	-.343		.500
.549	-.179	-.204	-.229	-.247	-.252	-.271	-.279	-.285	-.302	-.329		.549
.598	-.178	-.188	-.212	-.214	-.214	-.220	-.223	-.225	-.232	-.251		.598
.649	-.026	-.029	-.010	-.003	.006	.013	.014	.022	.019	.017		.649
.699	.179	.183	.206	.210	.218	.218	.219	.212	.214	.213		.699
.747	.288	.292	.308	.305	.306	.298	.301	.291	.290	.299		.747
.800	.358	.365	.374	.369	.375	.365	.358	.355	.355	.365		.800
.850	.404	.411	.418	.426	.424	.415	.415	.413	.405	.421		.850
.899	.430	.440	.455	.460	.451	.453	.463	.449	.450	.460		.899
.929	.426	.443	.461	.466	.472	.467	.465	.462	.466	.470		.929
.950	.415	.431	.451	.459	.465	.458	.461	.459	.455	.456		.950
.969	.371	.383	.405	.417	.420	.418	.421	.420	.414	.413		.969
.990	.244	.261	.284	.291	.297	.297	.301	.294	.284	.274		.990
.998	.090	.099	.125	.129	.134	.132	.138	.130	.117	.085		.998

[redacted]

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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					1.154	1.162	1.166	1.169	1.175	1.178		0.000
.003					.583	.626	.642	.672	.688	.715		.003
.012					-.311	-.220	-.178	-.159	-.135	-.114		.012
.019					-.560	-.478	-.450	-.434	-.387	-.350		.019
.028					-.645	-.620	-.598	-.531	-.525	-.474		.028
.047					-.679	-.672	-.639	-.613	-.575	-.561		.047
.072					-.691	-.647	-.626	-.587	-.562	-.526		.072
.098					-.667	-.721	-.695	-.665	-.646	-.618		.098
.150					-.558	-.598	-.663	-.669	-.642	-.615		.150
.200					-.495	-.451	-.555	-.629	-.623	-.621		.200
.250					-.476	-.479	-.446	-.567	-.595	-.609		.250
.301					-.455	-.475	-.497	-.411	-.558	-.548		.301
.351					-.434	-.446	-.444	-.427	-.347	-.537		.351
.398					-.417	-.433	-.436	-.423	-.395	-.339		.398
.448					-.421	-.438	-.439	-.430	-.414	-.353		.448
.499					-.445	-.457	-.458	-.455	-.448	-.415		.499
.549					-.464	-.471	-.475	-.471	-.460	-.437		.549
.600					-.511	-.524	-.531	-.526	-.513	-.498		.600
.652					-.521	-.544	-.559	-.560	-.555	-.541		.652
.700					-.565	-.592	-.609	-.603	-.603	-.595		.700
.750					-.604	-.664	-.673	-.678	-.669	-.664		.750
.801					-.579	-.665	-.759	-.771	-.772	-.758		.801
.850					-.489	-.470	-.459	-.572	-.633	-.613		.850
.899					-.331	-.301	-.276	-.241	-.220	-.216		.899
.932					-.201	-.174	-.159	-.131	-.130	-.135		.932
.949					-.127	-.105	-.091	-.080	-.090	-.107		.949
.972					-.032	-.024	-.023	-.027	-.049	-.079		.972
.981					-.004	-.005	-.010	-.014	-.030	-.060		.981
1.000					.022	.016	.014	.002	-.010	-.033		1.000
LOWER SURFACE												
.006					.397	.362	.362	.349	.384	.339		.006
.011					.083	.078	.070	.058	.051	.036		.011
.020					-.130	-.156	-.172	-.165	-.155	-.152		.020
.029					-.278	-.307	-.314	-.304	-.318	-.328		.029
.048					-.363	-.367	-.367	-.401	-.386	-.408		.048
.069					-.382	-.390	-.423	-.429	-.449	-.451		.069
.102					-.391	-.426	-.433	-.480	-.496	-.559		.102
.151					-.363	-.401	-.426	-.463	-.498	-.575		.151
.199					-.315	-.344	-.354	-.376	-.396	-.489		.199
.251					-.251	-.265	-.282	-.296	-.314	-.325		.251
.299					-.245	-.267	-.269	-.294	-.315	-.331		.299
.350					-.222	-.243	-.249	-.263	-.277	-.299		.350
.400					-.216	-.230	-.239	-.253	-.264	-.277		.400
.451					-.216	-.232	-.242	-.252	-.261	-.295		.451
.500					-.241	-.260	-.273	-.279	-.303	-.318		.500
.549					-.243	-.256	-.263	-.271	-.293	-.307		.549
.598					-.212	-.208	-.216	-.219	-.227	-.230		.598
.649					.005	.013	.007	.021	.022	.023		.649
.699					.225	.227	.223	.225	.222	.217		.699
.747					.324	.321	.313	.309	.305	.300		.747
.800					.396	.386	.383	.376	.372	.359		.800
.850					.440	.439	.437	.428	.430	.420		.850
.899					.476	.474	.470	.467	.465	.459		.899
.929					.486	.490	.482	.479	.472	.464		.929
.950					.472	.473	.466	.464	.461	.456		.950
.969					.425	.432	.422	.424	.416	.409		.969
.990					.294	.302	.294	.296	.283	.271		.990
.998					.126	.125	.117	.115	.105	.088		.998

[redacted]

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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					1.157	1.162	1.166	1.167	1.176			0.000
.003					.492	.537	.554	.574	.612			.003
.012					-.407	-.347	-.307	-.260	-.234			.012
.019					-.695	-.635	-.593	-.549	-.495			.019
.028					-.834	-.748	-.710	-.675	-.651			.028
.047					-.870	-.800	-.771	-.717	-.683			.047
.072					-.877	-.827	-.811	-.752	-.721			.072
.098					-.861	-.824	-.783	-.762	-.709			.098
.150					-.787	-.809	-.789	-.766	-.736			.150
.200					-.451	-.766	-.755	-.754	-.755			.200
.250					-.514	-.736	-.752	-.736	-.717			.250
.301					-.504	-.363	-.717	-.701	-.694			.301
.351					-.475	-.430	-.413	-.680	-.686			.351
.398					-.462	-.446	-.378	-.555	-.668			.398
.448					-.461	-.456	-.423	-.355	-.650			.448
.499					-.478	-.484	-.463	-.401	-.654			.499
.549					-.488	-.502	-.494	-.438	-.656			.549
.600					-.532	-.551	-.549	-.507	-.411			.600
.652					-.537	-.582	-.580	-.556	-.433			.652
.700					-.586	-.633	-.630	-.612	-.540			.700
.750					-.621	-.694	-.701	-.690	-.629			.750
.801					-.586	-.756	-.794	-.789	-.747			.801
.850					-.488	-.465	-.495	-.613	-.753			.850
.899					-.329	-.302	-.266	-.242	-.242			.899
.932					-.202	-.175	-.149	-.138	-.138			.932
.949					-.128	-.109	-.094	-.093	-.102			.949
.972					-.041	-.032	-.030	-.041	-.050			.972
.981					-.010	-.005	-.016	-.019	-.033			.981
1.000					.016	.013	.006	-.003	-.015			1.000
LOWER SURFACE												
.006					.497	.491	.491	.467	.480			.006
.011					.208	.210	.194	.202	.181			.011
.020					-.012	-.009	-.019	-.039	-.039			.020
.029					-.147	-.149	-.162	-.169	-.190			.029
.048					-.209	-.238	-.245	-.253	-.257			.048
.069					-.245	-.270	-.276	-.310	-.324			.069
.102					-.287	-.305	-.309	-.343	-.360			.102
.151					-.283	-.299	-.322	-.332	-.372			.151
.199					-.244	-.261	-.281	-.291	-.322			.199
.251					-.212	-.222	-.242	-.242	-.275			.251
.299					-.182	-.199	-.208	-.215	-.235			.299
.350					-.169	-.191	-.196	-.202	-.216			.350
.400					-.166	-.188	-.193	-.199	-.212			.400
.451					-.179	-.192	-.196	-.206	-.220			.451
.500					-.204	-.217	-.226	-.237	-.255			.500
.549					-.211	-.224	-.221	-.236	-.256			.549
.598					-.200	-.206	-.213	-.219	-.227			.598
.649					.014	.025	.025	.031	.023			.649
.699					.237	.240	.240	.241	.244			.699
.747					.333	.335	.340	.344	.332			.747
.800					.411	.409	.409	.410	.405			.800
.850					.459	.463	.457	.457	.461			.850
.899					.494	.495	.489	.500	.497			.899
.929					.497	.498	.496	.505	.494			.929
.950					.485	.489	.491	.489	.489			.950
.969					.433	.439	.437	.440	.438			.969
.990					.296	.302	.301	.302	.297			.990
.998					.120	.125	.116	.116	.103			.998

[redacted]

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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	1.027	1.062	1.114	1.139	1.147	1.158	1.159	1.171	1.173			0.000
.003	-.117	.044	.265	.355	.407	.463	.496	.531	.549			.003
.012	-1.178	-1.054	-.772	-.606	-.532	-.439	-.405	-.348	-.304			.012
.019	-1.201	-1.186	-1.159	-.983	-.875	-.808	-.726	-.680	-.611			.019
.028	-1.124	-1.168	-1.177	-1.042	-.965	-.858	-.832	-.767	-.716			.028
.047	-.916	-.952	-1.143	-1.024	-.947	-.865	-.827	-.780	-.746			.047
.072	-.757	-.809	-1.074	-1.064	-.997	-.918	-.886	-.828	-.803			.072
.098	-.645	-.687	-.702	-1.064	-1.001	-.940	-.911	-.843	-.797			.098
.150	-.570	-.601	-.658	-.947	-.969	-.908	-.882	-.820	-.787			.150
.200	-.517	-.544	-.595	-.477	-.948	-.895	-.872	-.834	-.804			.200
.250	-.480	-.504	-.547	-.536	-.712	-.868	-.837	-.810	-.782			.250
.301	-.447	-.465	-.512	-.527	-.445	-.821	-.811	-.772	-.754			.301
.351	-.424	-.444	-.485	-.493	-.455	-.823	-.804	-.790	-.757			.351
.398	-.404	-.422	-.464	-.482	-.462	-.439	-.770	-.767	-.742			.398
.448	-.402	-.417	-.462	-.477	-.477	-.382	-.752	-.757	-.741			.448
.499	-.404	-.430	-.475	-.486	-.493	-.426	-.459	-.761	-.743			.499
.549	-.400	-.426	-.472	-.500	-.514	-.476	-.394	-.756	-.746			.549
.600	-.427	-.454	-.499	-.533	-.557	-.543	-.464	-.775	-.771			.600
.652	-.417	-.442	-.505	-.544	-.559	-.572	-.524	-.785	-.786			.652
.700	-.436	-.462	-.528	-.572	-.612	-.627	-.599	-.781	-.818			.700
.750	-.448	-.477	-.541	-.591	-.646	-.701	-.680	-.598	-.859			.750
.801	-.444	-.477	-.529	-.550	-.606	-.785	-.786	-.529	-.695			.801
.850	-.428	-.453	-.483	-.490	-.499	-.468	-.521	-.398	-.325			.850
.899	-.348	-.365	-.356	-.342	-.331	-.302	-.271	-.235	-.200			.899
.932	-.270	-.262	-.242	-.214	-.203	-.173	-.152	-.129	-.119			.932
.949	-.200	-.194	-.160	-.140	-.128	-.108	-.094	-.073	-.097			.949
.972	-.099	-.086	-.061	-.043	-.036	-.027	-.022	-.021	-.059			.972
.981	-.067	-.053	-.034	-.026	-.014	-.004	-.011	.000	-.028			.981
1.000	-.011	-.007	.005	.010	.011	.014	.023	.018	-.024			1.000
LOWER SURFACE												
.006	.635	.634	.618	.598	.592	.568	.597	.553	.525			.006
.011	.380	.372	.338	.327	.318	.309	.293	.281	.264			.011
.020	.194	.163	.137	.116	.120	.090	.091	.069	.048			.020
.029	.058	.020	.009	-.017	-.039	-.047	-.061	-.071	-.094			.029
.048	-.007	-.031	-.084	-.108	-.106	-.134	-.129	-.143	-.168			.048
.069	-.064	-.084	-.128	-.147	-.164	-.181	-.192	-.204	-.223			.069
.102	-.110	-.143	-.158	-.196	-.207	-.231	-.239	-.259	-.262			.102
.151	-.121	-.139	-.179	-.190	-.210	-.231	-.232	-.261	-.284			.151
.199	-.111	-.125	-.152	-.178	-.178	-.206	-.206	-.231	-.251			.199
.251	-.108	-.126	-.154	-.169	-.174	-.195	-.192	-.209	-.227			.251
.299	-.084	-.090	-.110	-.123	-.135	-.147	-.149	-.161	-.173			.299
.350	-.081	-.091	-.113	-.131	-.137	-.144	-.142	-.156	-.180			.350
.400	-.086	-.096	-.113	-.126	-.134	-.142	-.145	-.159	-.178			.400
.451	-.091	-.108	-.123	-.148	-.146	-.159	-.158	-.172	-.189			.451
.500	-.123	-.130	-.153	-.174	-.177	-.191	-.193	-.205	-.216			.500
.549	-.129	-.146	-.168	-.180	-.184	-.204	-.208	-.211	-.234			.549
.598	-.149	-.159	-.177	-.188	-.191	-.189	-.195	-.205	-.206			.598
.649	.003	.006	.015	.024	.022	.025	.035	.037	.035			.649
.699	.203	.218	.231	.239	.247	.254	.254	.255	.256			.699
.747	.304	.325	.340	.346	.355	.359	.354	.359	.355			.747
.800	.372	.395	.414	.422	.429	.430	.433	.427	.428			.800
.850	.424	.440	.465	.476	.483	.480	.480	.484	.481			.850
.899	.444	.466	.493	.501	.509	.511	.515	.508	.515			.899
.929	.449	.465	.494	.501	.513	.515	.519	.519	.515			.929
.950	.431	.456	.477	.491	.501	.496	.509	.506	.503			.950
.969	.379	.403	.426	.434	.447	.450	.452	.456	.439			.969
.990	.249	.263	.284	.297	.305	.305	.312	.312	.299			.990
.998	.083	.093	.106	.105	.116	.124	.120	.127	.109			.998

 [redacted]

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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	.963	1.026	1.103	1.129	1.138	1.147	1.156	1.170	1.179			0.000
.003	-.320	-.114	.160	.287	.319	.393	.431	.467	.502			.003
.012	-1.422	-1.261	-.889	-.689	-.608	-.521	-.463	-.422	-.366			.012
.019	-1.356	-1.521	-1.330	-1.127	-1.020	-.924	-.852	-.790	-.728			.019
.028	-1.272	-1.380	-1.367	-1.181	-1.062	-.964	-.913	-.851	-.817			.028
.047	-1.074	-1.133	-1.330	-1.129	-1.052	-.967	-.922	-.865	-.814			.047
.072	-.861	-.896	-1.321	-1.177	-1.100	-1.020	-.967	-.903	-.868			.072
.098	-.720	-.773	-1.223	-1.166	-1.088	-1.014	-.974	-.932	-.879			.098
.150	-.622	-.670	-.623	-1.103	-1.067	-1.010	-.968	-.921	-.871			.150
.200	-.558	-.593	-.621	-1.040	-1.038	-.989	-.944	-.913	-.867			.200
.250	-.514	-.544	-.580	-.400	-1.014	-.977	-.949	-.914	-.858			.250
.301	-.475	-.512	-.548	-.466	-.951	-.930	-.882	-.868	-.807			.301
.351	-.451	-.470	-.511	-.473	-.885	-.898	-.887	-.853	-.801			.351
.398	-.431	-.458	-.489	-.472	-.367	-.897	-.878	-.848	-.813			.398
.448	-.424	-.445	-.483	-.488	-.395	-.867	-.859	-.832	-.799			.448
.499	-.423	-.449	-.489	-.506	-.451	-.873	-.863	-.837	-.805			.499
.549	-.432	-.452	-.483	-.517	-.491	-.804	-.855	-.828	-.808			.549
.600	-.442	-.466	-.510	-.556	-.547	-.411	-.862	-.847	-.830			.600
.652	-.437	-.460	-.519	-.554	-.571	-.445	-.885	-.866	-.848			.652
.700	-.454	-.479	-.529	-.585	-.617	-.516	-.820	-.902	-.888			.700
.750	-.459	-.485	-.548	-.601	-.644	-.603	-.443	-.948	-.925			.750
.801	-.455	-.477	-.525	-.567	-.607	-.653	-.496	-.499	-.509			.801
.850	-.432	-.452	-.475	-.492	-.497	-.485	-.431	-.304	-.298			.850
.899	-.353	-.356	-.348	-.344	-.338	-.320	-.270	-.193	-.196			.899
.932	-.266	-.262	-.232	-.214	-.209	-.190	-.139	-.125	-.149			.932
.949	-.208	-.191	-.162	-.143	-.130	-.116	-.080	-.092	-.127			.949
.972	-.100	-.090	-.061	-.049	-.042	-.024	-.014	-.047	-.094			.972
.981	-.071	-.052	-.031	-.016	-.008	.001	.004	-.038	-.087			.981
1.000	-.010	-.006	.008	.011	.016	.028	.026	-.013	-.071			1.000
LOWER SURFACE												
.006	.734	.736	.693	.688	.676	.669	.656	.645	.623			.006
.011	.495	.482	.449	.427	.424	.401	.383	.369	.343			.011
.020	.292	.263	.258	.219	.204	.195	.177	.171	.133			.020
.029	.125	.153	.093	.083	.072	.067	.049	.028	.004			.029
.048	.058	.055	.003	-.001	-.018	-.037	-.044	-.062	-.074			.048
.069	.017	-.019	-.038	-.070	-.071	-.078	-.089	-.107	-.141			.069
.102	-.053	-.070	-.092	-.117	-.126	-.135	-.150	-.187	-.200			.102
.151	-.079	-.090	-.113	-.129	-.153	-.156	-.165	-.180	-.218			.151
.199	-.078	-.080	-.112	-.126	-.128	-.138	-.146	-.153	-.195			.199
.251	-.078	-.086	-.109	-.127	-.118	-.126	-.137	-.141	-.175			.251
.299	-.062	-.064	-.075	-.084	-.091	-.093	-.101	-.114	-.152			.299
.350	-.059	-.064	-.079	-.090	-.095	-.098	-.110	-.115	-.156			.350
.400	-.062	-.066	-.088	-.096	-.100	-.106	-.108	-.121	-.160			.400
.451	-.072	-.083	-.096	-.107	-.112	-.122	-.120	-.131	-.171			.451
.500	-.103	-.114	-.133	-.147	-.144	-.142	-.153	-.168	-.207			.500
.549	-.114	-.125	-.145	-.158	-.153	-.156	-.163	-.188	-.220			.549
.598	-.138	-.145	-.151	-.159	-.164	-.157	-.158	-.179	-.199			.598
.649	.016	.019	.028	.032	.042	.052	.052	.044	.038			.649
.699	.208	.223	.240	.259	.262	.267	.276	.264	.253			.699
.747	.313	.329	.354	.358	.367	.375	.371	.367	.357			.747
.800	.379	.397	.418	.431	.433	.439	.442	.440	.430			.800
.850	.423	.442	.464	.479	.486	.491	.489	.488	.475			.850
.899	.446	.471	.491	.505	.515	.524	.521	.522	.511			.899
.929	.445	.467	.494	.506	.520	.525	.527	.520	.506			.929
.950	.433	.452	.483	.494	.501	.512	.513	.505	.493			.950
.969	.380	.399	.428	.438	.453	.455	.465	.449	.435			.969
.990	.247	.263	.285	.298	.306	.322	.322	.304	.282			.990
.998	.074	.090	.104	.117	.129	.138	.135	.105	.070			.998

[redacted]

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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	.917	.981	1.072	1.112	1.130	1.140	1.149	1.162	1.164			0.000
.003	-.494	-.268	.092	.195	.253	.342	.354	.406	.451			.003
.012	-1.680	-1.462	-.955	-.762	-.678	-.587	-.551	-.474	-.433			.012
.019	-1.629	-1.797	-1.455	-1.216	-1.117	-.984	-.929	-.865	-.813			.019
.028	-1.454	-1.632	-1.494	-1.282	-1.179	-1.059	-1.003	-.944	-.885			.028
.047	-1.001	-1.013	-1.455	-1.272	-1.149	-1.042	-.981	-.924	-.850			.047
.072	-.949	-1.034	-1.459	-1.272	-1.175	-1.088	-1.034	-.983	-.910			.072
.098	-.802	-.848	-1.381	-1.274	-1.169	-1.082	-1.029	-.977	-.913			.098
.150	-.677	-.720	-1.195	-1.213	-1.162	-1.075	-1.018	-.990	-.919			.150
.200	-.608	-.641	-.572	-1.195	-1.121	-1.043	-.995	-.967	-.910			.200
.250	-.549	-.589	-.578	-1.121	-1.108	-1.043	-.990	-.947	-.920			.250
.301	-.505	-.543	-.563	-1.034	-1.055	-1.000	-.959	-.922	-.875			.301
.351	-.477	-.507	-.532	-.404	-1.028	-.986	-.945	-.907	-.864			.351
.398	-.456	-.481	-.512	-.403	-.996	-.961	-.917	-.894	-.860			.398
.448	-.446	-.472	-.505	-.449	-.929	-.951	-.914	-.881	-.861			.448
.499	-.444	-.467	-.505	-.486	-.446	-.948	-.918	-.885	-.860			.499
.549	-.444	-.465	-.503	-.504	-.397	-.948	-.929	-.893	-.868			.549
.600	-.458	-.486	-.536	-.554	-.464	-.952	-.936	-.912	-.874			.600
.652	-.446	-.473	-.524	-.566	-.505	-.968	-.948	-.921	-.899			.652
.700	-.460	-.489	-.540	-.593	-.572	-.916	-.982	-.959	-.931			.700
.750	-.465	-.497	-.549	-.604	-.613	-.471	-.980	-.997	-.883			.750
.801	-.454	-.484	-.536	-.573	-.594	-.418	-.427	-.429	-.383			.801
.850	-.432	-.451	-.478	-.496	-.502	-.384	-.302	-.299	-.297			.850
.899	-.356	-.359	-.356	-.357	-.342	-.257	-.190	-.203	-.234			.899
.932	-.268	-.261	-.231	-.227	-.213	-.157	-.117	-.165	-.206			.932
.949	-.201	-.189	-.166	-.150	-.140	-.098	-.079	-.144	-.182			.949
.972	-.102	-.087	-.065	-.049	-.034	-.019	-.045	-.112	-.165			.972
.981	-.065	-.051	-.027	-.018	-.002	.006	-.039	-.105	-.155			.981
1.000	-.007	-.007	.002	.016	.032	.025	.009	-.073	-.126			1.000
LOWER SURFACE												
.006	.821	.820	.800	.759	.752	.744	.722	.696	.682			.006
.011	.583	.577	.523	.523	.514	.474	.473	.447	.403			.011
.020	.384	.371	.319	.298	.301	.281	.272	.231	.210			.020
.029	.241	.216	.186	.172	.166	.124	.110	.088	.064			.029
.048	.141	.120	.090	.069	.073	.048	.032	.009	-.017			.048
.069	.066	.067	.018	.014	.007	-.011	-.036	-.054	-.104			.069
.102	.006	-.001	-.038	-.053	-.054	-.067	-.082	-.102	-.157			.102
.151	-.026	-.038	-.072	-.070	-.087	-.092	-.114	-.132	-.179			.151
.199	-.038	-.043	-.060	-.079	-.071	-.087	-.106	-.114	-.166			.199
.251	-.035	-.037	-.068	-.068	-.079	-.092	-.107	-.122	-.163			.251
.299	-.020	-.025	-.039	-.047	-.044	-.047	-.071	-.095	-.121			.299
.350	-.025	-.031	-.049	-.053	-.052	-.066	-.075	-.094	-.133			.350
.400	-.033	-.038	-.062	-.056	-.059	-.069	-.092	-.103	-.136			.400
.451	-.052	-.053	-.075	-.073	-.080	-.089	-.118	-.128	-.159			.451
.500	-.081	-.094	-.107	-.122	-.109	-.135	-.144	-.172	-.195			.500
.549	-.093	-.097	-.125	-.128	-.128	-.132	-.149	-.185	-.204			.549
.598	-.121	-.120	-.136	-.135	-.133	-.136	-.143	-.179	-.208			.598
.649	.029	.036	.040	.047	.060	.062	.061	.038	.024			.649
.699	.219	.236	.254	.262	.276	.281	.276	.269	.255			.699
.747	.326	.339	.357	.373	.381	.381	.376	.368	.353			.747
.800	.395	.412	.429	.444	.454	.454	.443	.433	.427			.800
.850	.431	.458	.472	.495	.500	.509	.497	.483	.478			.850
.899	.458	.477	.503	.522	.529	.533	.529	.509	.502			.899
.929	.451	.479	.497	.519	.531	.534	.527	.510	.505			.929
.950	.439	.458	.486	.507	.516	.521	.513	.490	.482			.950
.969	.383	.403	.426	.445	.461	.464	.454	.436	.419			.969
.990	.247	.268	.289	.303	.323	.324	.304	.274	.249			.990
.998	.084	.097	.109	.118	.138	.141	.099	.052	.021			.998

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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	.834	.916	1.042	1.079	1.109	1.129	1.133					0.000
.003	-.713	-.405	-.039	.107	.203	.253	.314					.003
.012	-2.034	-1.650	-1.072	-.858	-.756	-.652	-.605					.012
.019	-1.848	-2.219	-1.552	-1.314	-1.198	-1.090	-1.011					.019
.028	-1.642	-2.059	-1.639	-1.393	-1.251	-1.153	-1.065					.028
.047	-1.108	-1.330	-1.579	-1.365	-1.275	-1.137	-1.070					.047
.072	-1.028	-1.080	-1.565	-1.379	-1.281	-1.163	-1.100					.072
.098	-.867	-.911	-1.534	-1.364	-1.254	-1.158	-1.111					.098
.150	-.743	-.785	-1.477	-1.318	-1.222	-1.157	-1.084					.150
.200	-.651	-.685	-.871	-1.285	-1.208	-1.126	-1.072					.200
.250	-.589	-.626	-.473	-1.237	-1.186	-1.111	-1.045					.250
.301	-.551	-.580	-.527	-1.222	-1.152	-1.071	-1.034					.301
.351	-.510	-.529	-.519	-1.174	-1.108	-1.037	-.999					.351
.398	-.486	-.504	-.508	-.762	-1.089	-1.023	-.991					.398
.448	-.472	-.494	-.512	-.406	-1.072	-1.017	-.969					.448
.499	-.470	-.487	-.526	-.395	-1.058	-1.013	-.967					.499
.549	-.460	-.482	-.520	-.439	-1.017	-1.000	-.969					.549
.600	-.470	-.498	-.542	-.500	-.507	-1.020	-.997					.600
.652	-.462	-.480	-.536	-.523	-.415	-1.029	-1.002					.652
.700	-.472	-.496	-.554	-.568	-.440	-1.058	-1.032					.700
.750	-.472	-.506	-.550	-.584	-.504	-.672	-.730					.750
.801	-.467	-.485	-.549	-.561	-.512	-.418	-.418					.801
.850	-.445	-.451	-.481	-.499	-.472	-.304	-.299					.850
.899	-.352	-.351	-.357	-.356	-.333	-.200	-.211					.899
.932	-.269	-.244	-.233	-.229	-.203	-.120	-.170					.932
.949	-.202	-.182	-.153	-.156	-.132	-.083	-.138					.949
.972	-.097	-.082	-.063	-.045	-.030	-.035	-.113					.972
.981	-.068	-.049	-.034	-.012	.001	-.007	-.089					.981
1.000	-.020	-.008	.002	.018	.040	.015	-.077					1.000
LOWER SURFACE												
.006	.892	.887	.845	.836	.812	.789	.774					.006
.011	.680	.655	.628	.603	.569	.562	.523					.011
.020	.490	.454	.417	.393	.383	.362	.338					.020
.029	.316	.330	.289	.262	.247	.226	.198					.029
.048	.200	.204	.180	.162	.137	.131	.085					.048
.069	.138	.117	.085	.085	.088	.061	.028					.069
.102	.051	.058	.022	.020	.022	-.004	-.027					.102
.151	.013	.007	-.016	-.015	-.023	-.045	-.057					.151
.199	.011	-.004	-.018	-.024	-.036	-.033	-.068					.199
.251	-.008	-.007	-.027	-.032	-.032	-.054	-.066					.251
.299	.008	.009	-.002	.000	-.001	-.019	-.043					.299
.350	-.003	-.002	-.016	-.014	-.015	-.022	-.054					.350
.400	-.008	-.015	-.028	-.032	-.031	-.041	-.066					.400
.451	-.032	-.036	-.043	-.050	-.043	-.058	-.077					.451
.500	-.060	-.068	-.085	-.082	-.084	-.084	-.125					.500
.549	-.074	-.086	-.105	-.096	-.092	-.115	-.150					.549
.598	-.096	-.104	-.117	-.108	-.100	-.124	-.153					.598
.649	.047	.045	.055	.067	.078	.068	.057					.649
.699	.229	.247	.264	.279	.286	.289	.269					.699
.747	.332	.346	.372	.380	.393	.388	.382					.747
.800	.401	.414	.440	.454	.464	.459	.439					.800
.850	.439	.458	.478	.500	.508	.506	.495					.850
.899	.458	.482	.509	.523	.529	.531	.522					.899
.929	.461	.476	.511	.523	.533	.532	.521					.929
.950	.445	.460	.495	.513	.526	.518	.500					.950
.969	.391	.408	.438	.454	.470	.459	.440					.969
.990	.249	.269	.291	.315	.330	.313	.277					.990
.998	.083	.092	.106	.127	.149	.129	.074					.998

TABLE II. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 12 - 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	.667	.817	.970	1.024	1.063							0.000
.003	-1.116	-.662	-.221	-.020	.054							.003
.012	-2.674	-1.948	-1.279	-1.044	-.934							.012
.019	-2.577	-2.452	-1.675	-1.445	-1.317							.019
.028	-1.922	-2.517	-1.772	-1.509	-1.402							.028
.047	-1.420	-2.331	-1.756	-1.514	-1.410							.047
.072	-1.251	-1.783	-1.757	-1.532	-1.411							.072
.098	-1.039	-.909	-1.710	-1.501	-1.374							.098
.150	-.854	-.844	-1.661	-1.464	-1.368							.150
.200	-.747	-.763	-1.598	-1.413	-1.336							.200
.250	-.669	-.683	-1.536	-1.384	-1.316							.250
.301	-.611	-.630	-.858	-1.353	-1.278							.301
.351	-.567	-.580	-.544	-1.327	-1.243							.351
.398	-.527	-.545	-.405	-1.297	-1.235							.398
.448	-.516	-.528	-.430	-1.283	-1.220							.448
.499	-.503	-.516	-.462	-1.260	-1.210							.499
.549	-.487	-.508	-.487	-1.146	-1.189							.549
.600	-.502	-.518	-.528	-.550	-1.207							.600
.652	-.487	-.504	-.530	-.404	-1.206							.652
.700	-.493	-.510	-.541	-.408	-.682							.700
.750	-.501	-.504	-.552	-.445	-.482							.750
.801	-.480	-.472	-.527	-.452	-.388							.801
.850	-.440	-.433	-.480	-.414	-.312							.850
.899	-.352	-.321	-.362	-.308	-.220							.899
.932	-.253	-.225	-.247	-.202	-.137							.932
.949	-.190	-.163	-.174	-.145	-.092							.949
.972	-.091	-.072	-.065	-.037	-.019							.972
.981	-.066	-.050	-.025	-.002	-.016							.981
1.000	-.019	-.010	.019	.036	.032							1.000
LOWER SURFACE												
.006	.984	.978	.957	.939	.917							.006
.011	.807	.807	.772	.725	.709							.011
.020	.631	.600	.567	.540	.518							.020
.029	.466	.467	.410	.414	.384							.029
.048	.357	.320	.299	.281	.263							.048
.069	.253	.237	.219	.209	.204							.069
.102	.161	.160	.139	.138	.118							.102
.151	.106	.095	.084	.083	.068							.151
.199	.081	.076	.085	.076	.064							.199
.251	.064	.048	.071	.067	.042							.251
.299	.069	.057	.055	.046	.052							.299
.350	.055	.042	.044	.038	.045							.350
.400	.034	.030	.028	.031	.023							.400
.451	.018	.006	.008	.010	.008							.451
.500	-.023	-.023	-.030	-.026	-.034							.500
.549	-.042	-.045	-.048	-.047	-.051							.549
.598	-.051	-.060	-.047	-.047	-.057							.598
.649	.064	.067	.087	.103	.103							.649
.699	.246	.259	.289	.304	.305							.699
.747	.345	.364	.392	.407	.411							.747
.800	.407	.424	.456	.472	.479							.800
.850	.449	.471	.496	.512	.521							.850
.899	.468	.487	.520	.537	.548							.899
.929	.466	.485	.518	.537	.544							.929
.950	.450	.466	.504	.528	.534							.950
.969	.393	.413	.446	.464	.469							.969
.990	.256	.267	.306	.331	.327							.990
.998	.073	.079	.128	.149	.121							.998

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 26a

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

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	1.066	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	-.080	-.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	-.368	.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	-.418	-.436	-.435	-.446	-.446	-.443	-.433	.200
.250	-.325	-.335	-.365	-.408	-.419	-.438	-.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	-.568	.600
.652	-.331	-.355	-.385	-.414	-.436	-.464	-.489	-.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

[redacted]

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL 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					-.483	-.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

[redacted]

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL 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	-.656	-.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	-.589	.200
.250					-.499	-.553	-.605	-.615	-.623	-.612	-.592	.250
.301					-.499	-.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	-.484	-.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

[redacted]

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL 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	.080	.259	.376	.408	.452	.481	.506	.536	.559	.580	.003
.012	-1.043	-.980	-.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	.398
.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	-.438	-.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	-.158	-.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

[redacted]

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL 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	-.508	-.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

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 26a - 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	.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	-.661	-.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	-.881	-.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	.256	.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	-.178	-.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

[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL 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	-.880	-.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	-.389	-.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	.080	.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

[redacted]

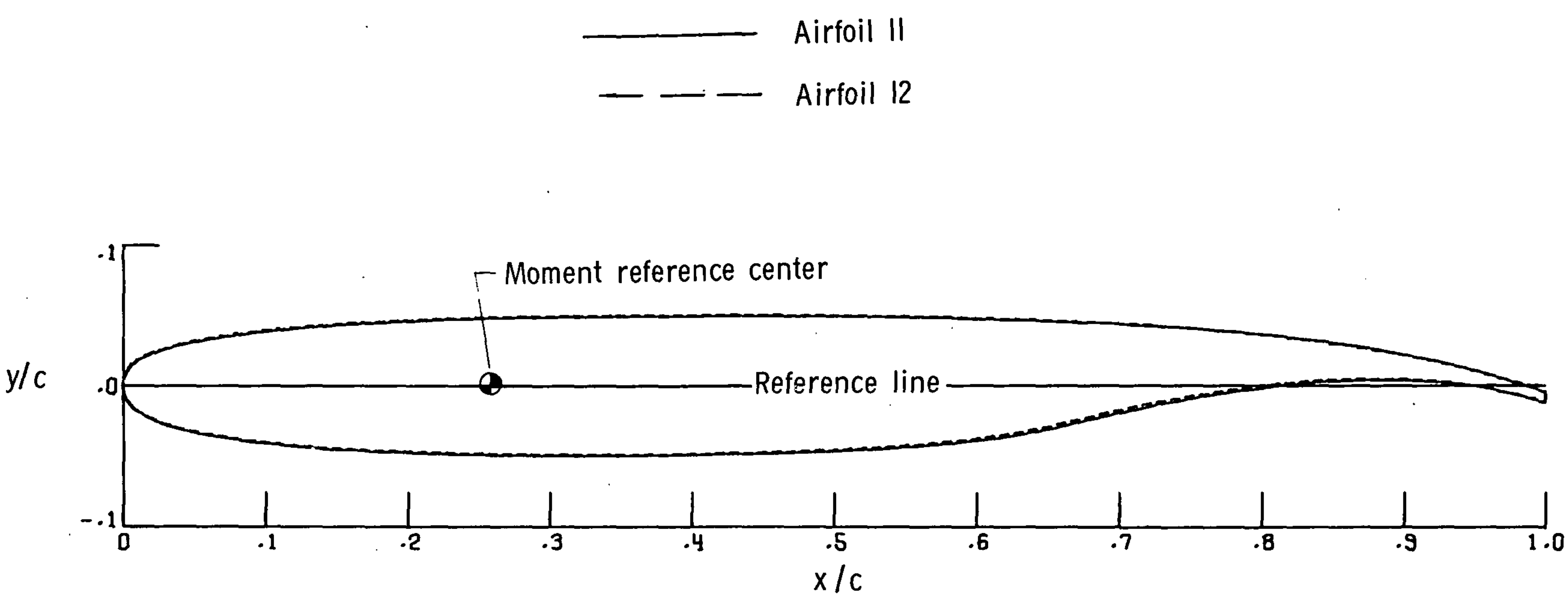
[redacted]

TABLE III. - SURFACE PRESSURE DISTRIBUTIONS FOR SUPERCRITICAL AIRFOIL 26a - 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	.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	-.046							.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

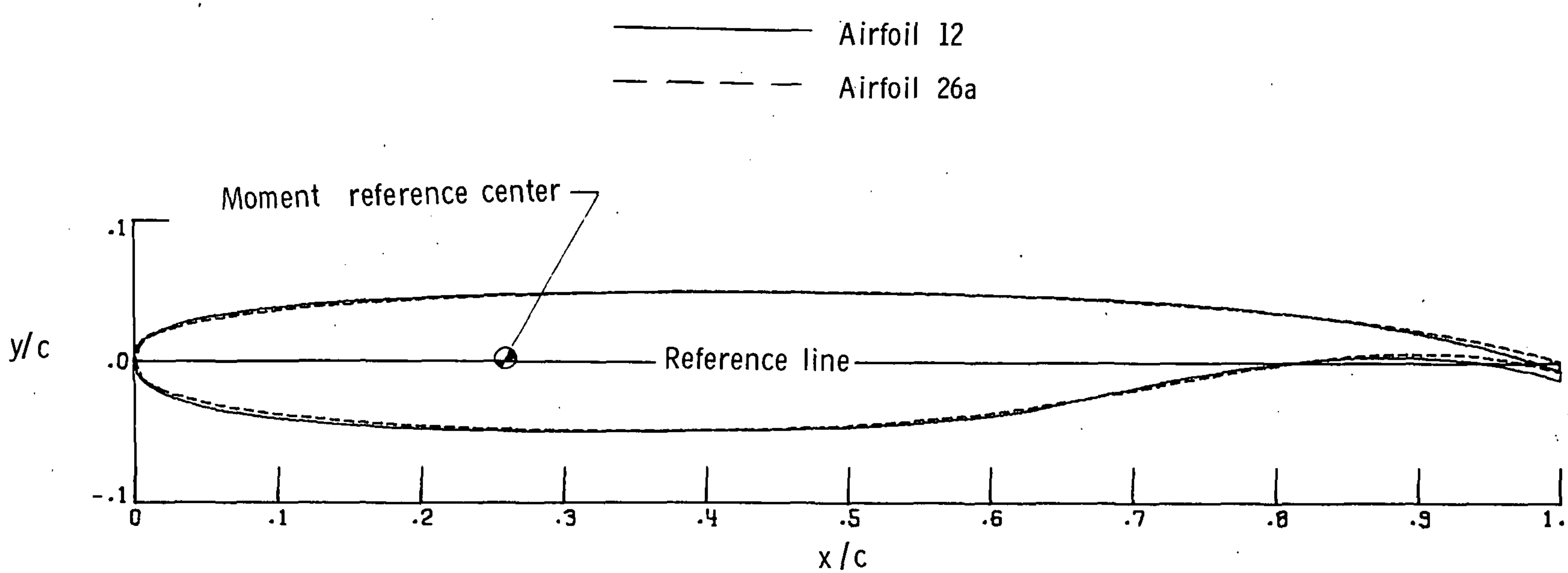
[redacted]



(a) Airfoils 11 and 12.

Figure 1.- Airfoil sketches.

[redacted]

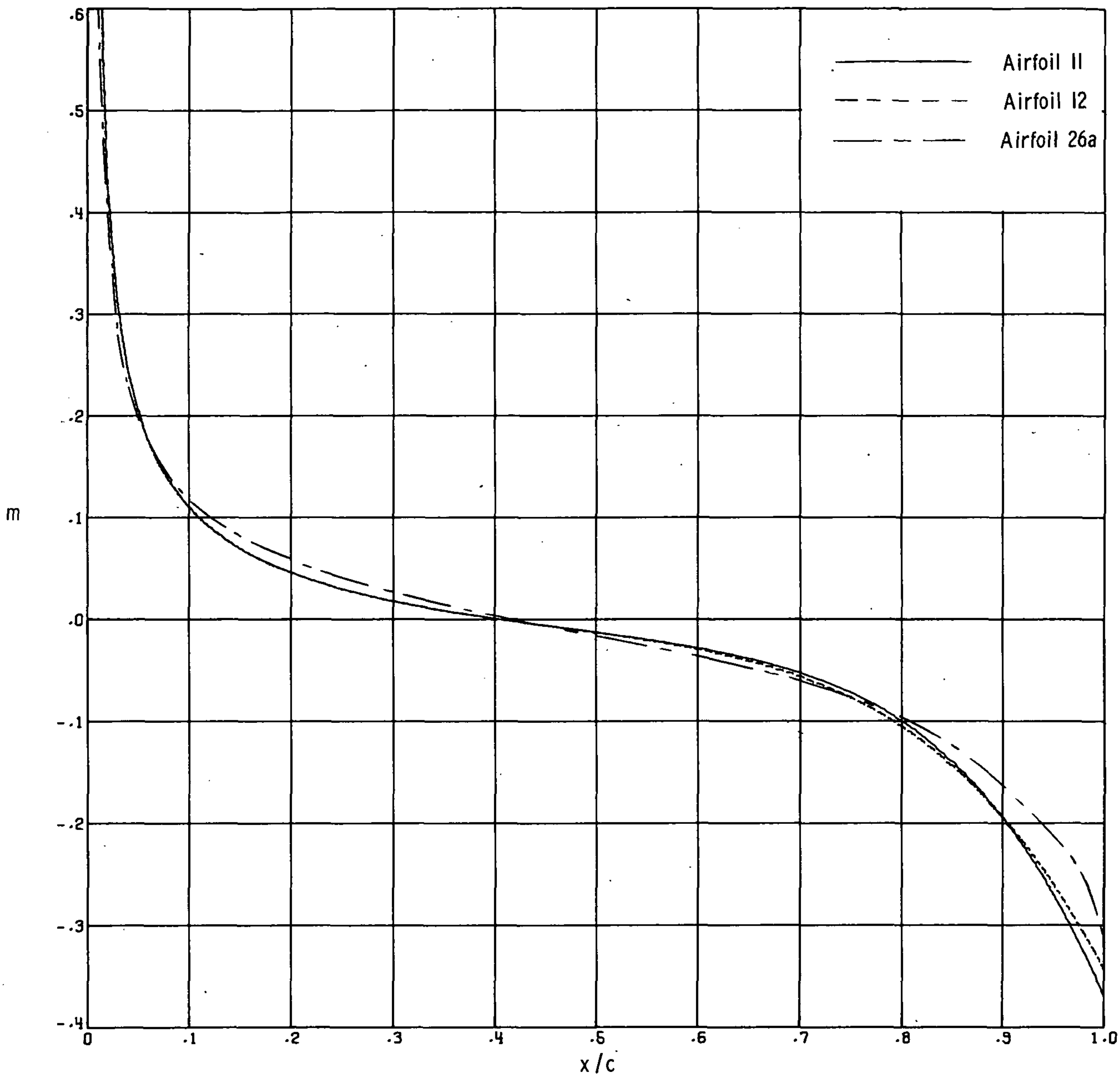


(b) Airfoils 12 and 26a.

Figure 1.- Concluded.

[redacted]

[redacted]

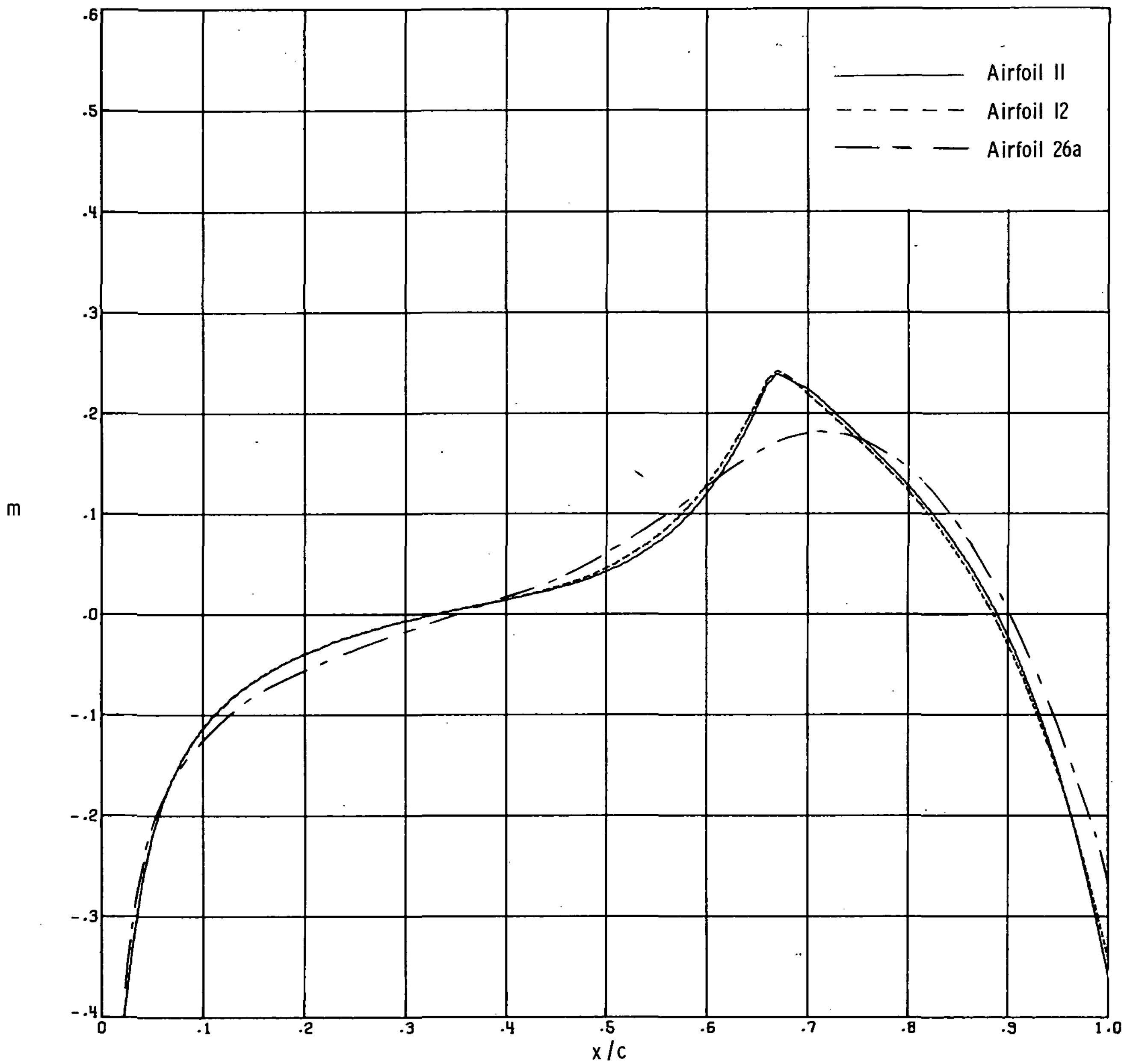


(a) Upper surface.

Figure 2.- Chordwise distribution of airfoil surface slopes.

[redacted]

[redacted]

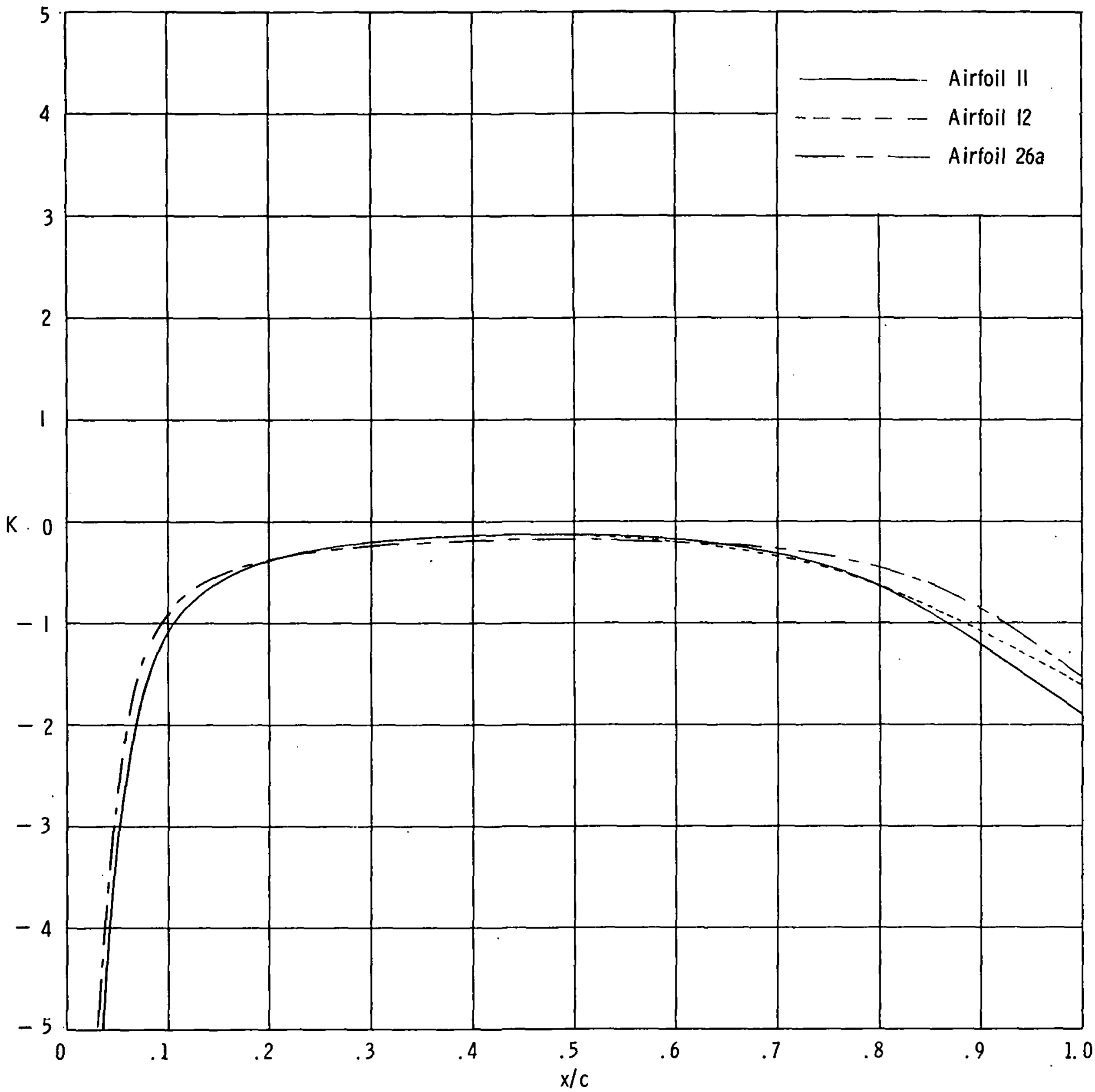


(b) Lower surface.

Figure 2.- Concluded.

[redacted]

[redacted]

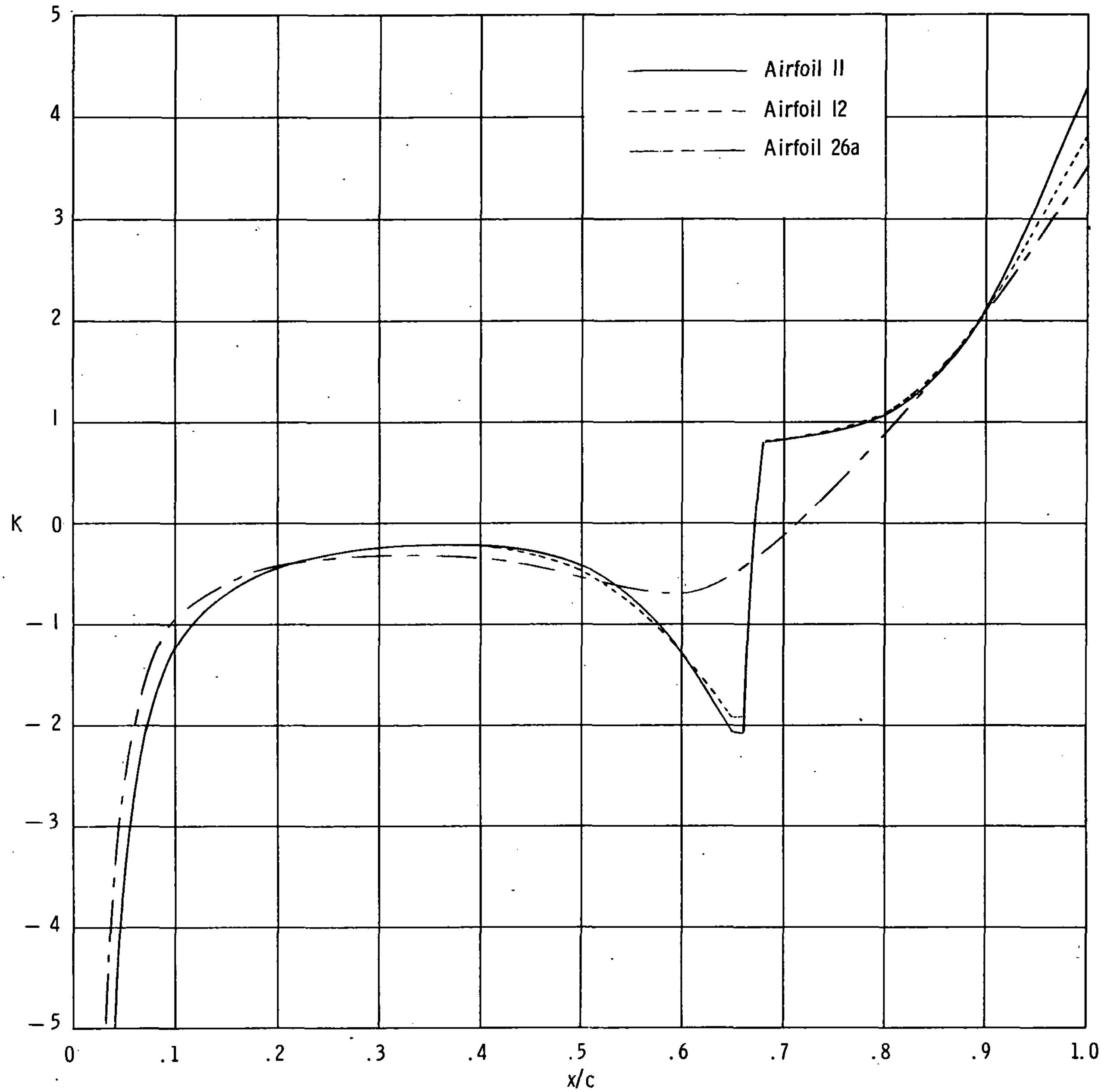


(a) Upper surface.

Figure 3.- Chordwise distribution of airfoil surface curvatures.

[redacted]

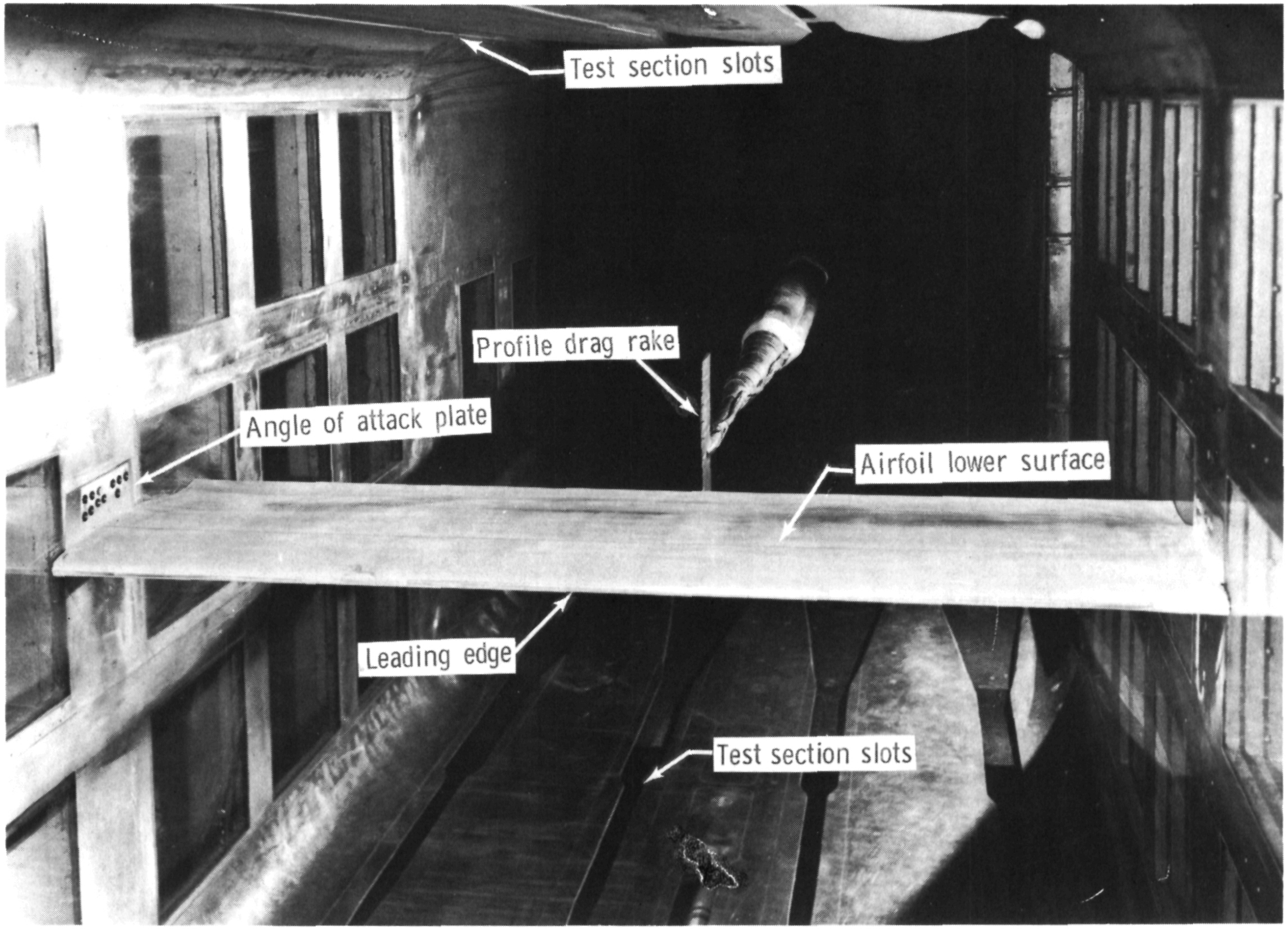
[redacted]



(b) Lower surface.

Figure 3.- Concluded.

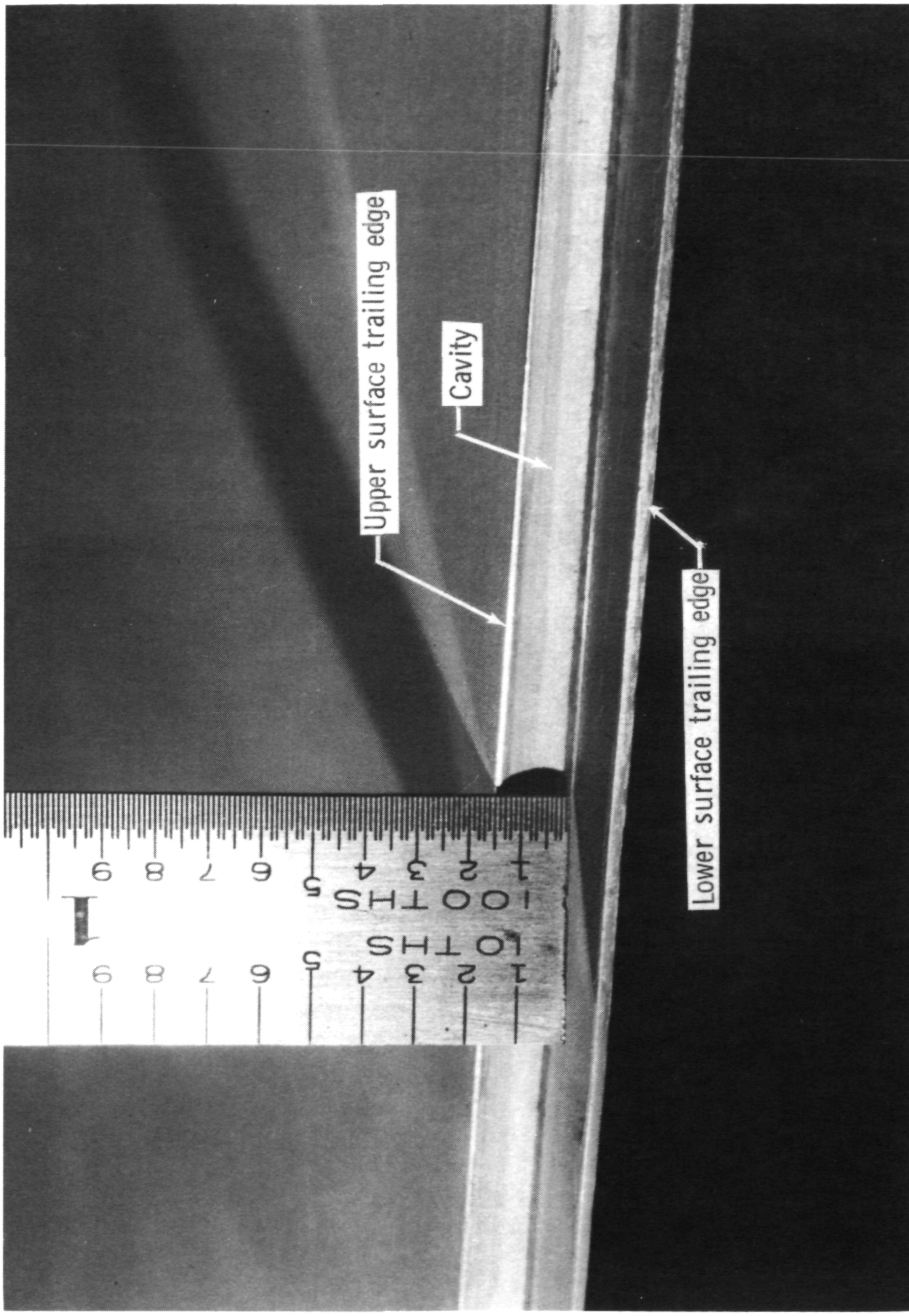
[redacted]



L-73-1225.1

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

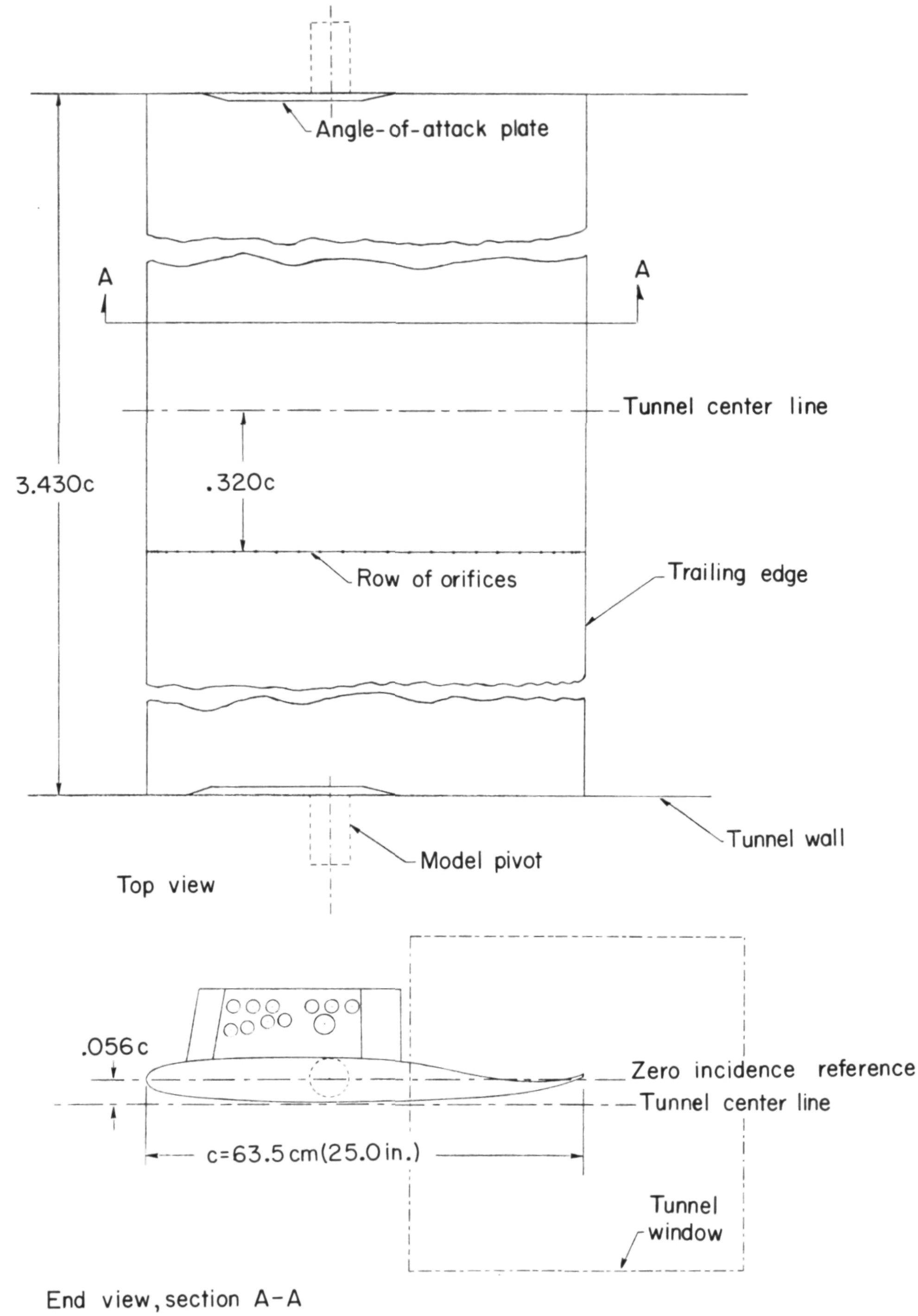
Figure 4.- Photographs of model in tunnel.



L-73-1227.1

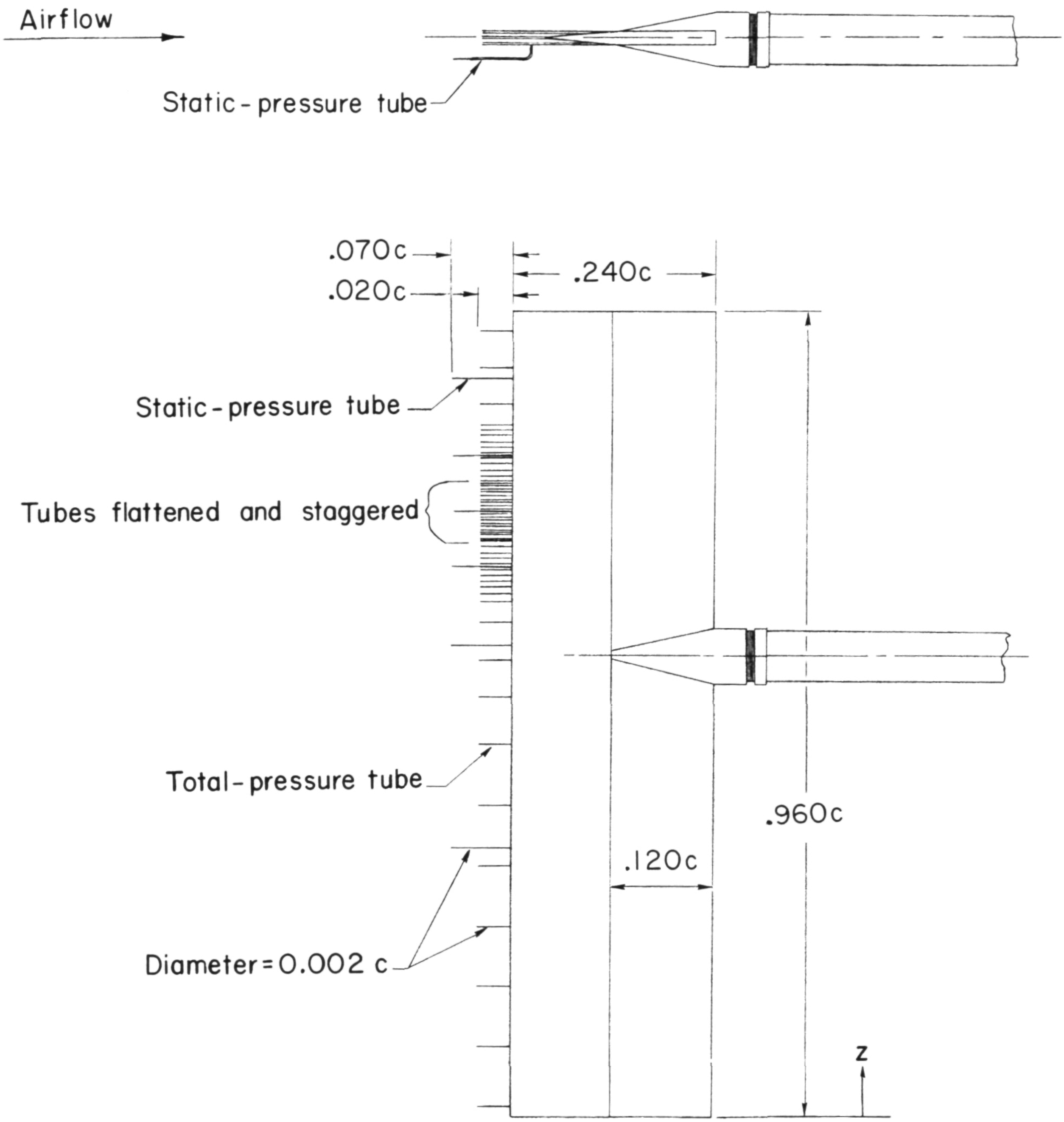
(b) Trailing-edge cavity.

Figure 4.- Concluded.



(a) Airfoil mounted in tunnel.

Figure 5.- Apparatus. Dimensions in terms of chord (c = 63.5 cm (25.0 in.)).



(b) Profile drag rake.

Figure 5.- Concluded.

[redacted]

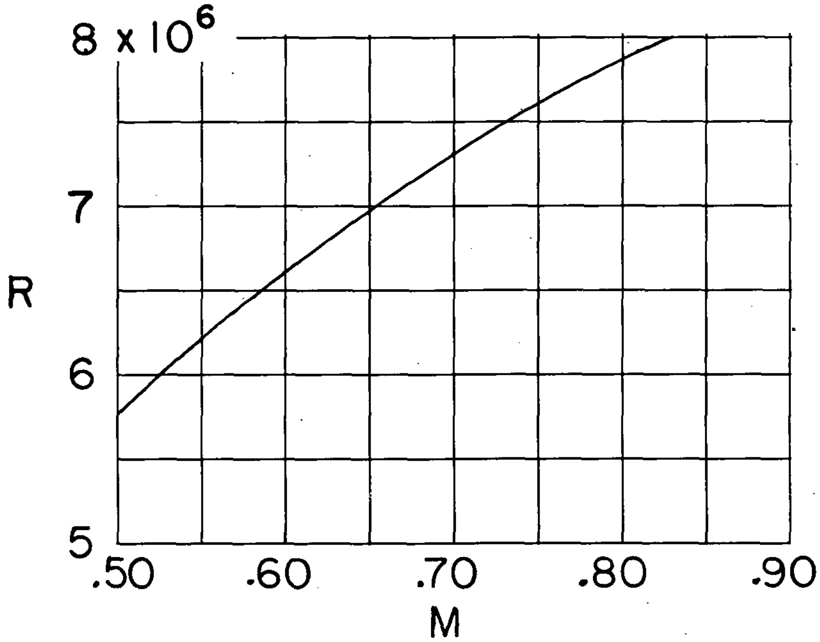
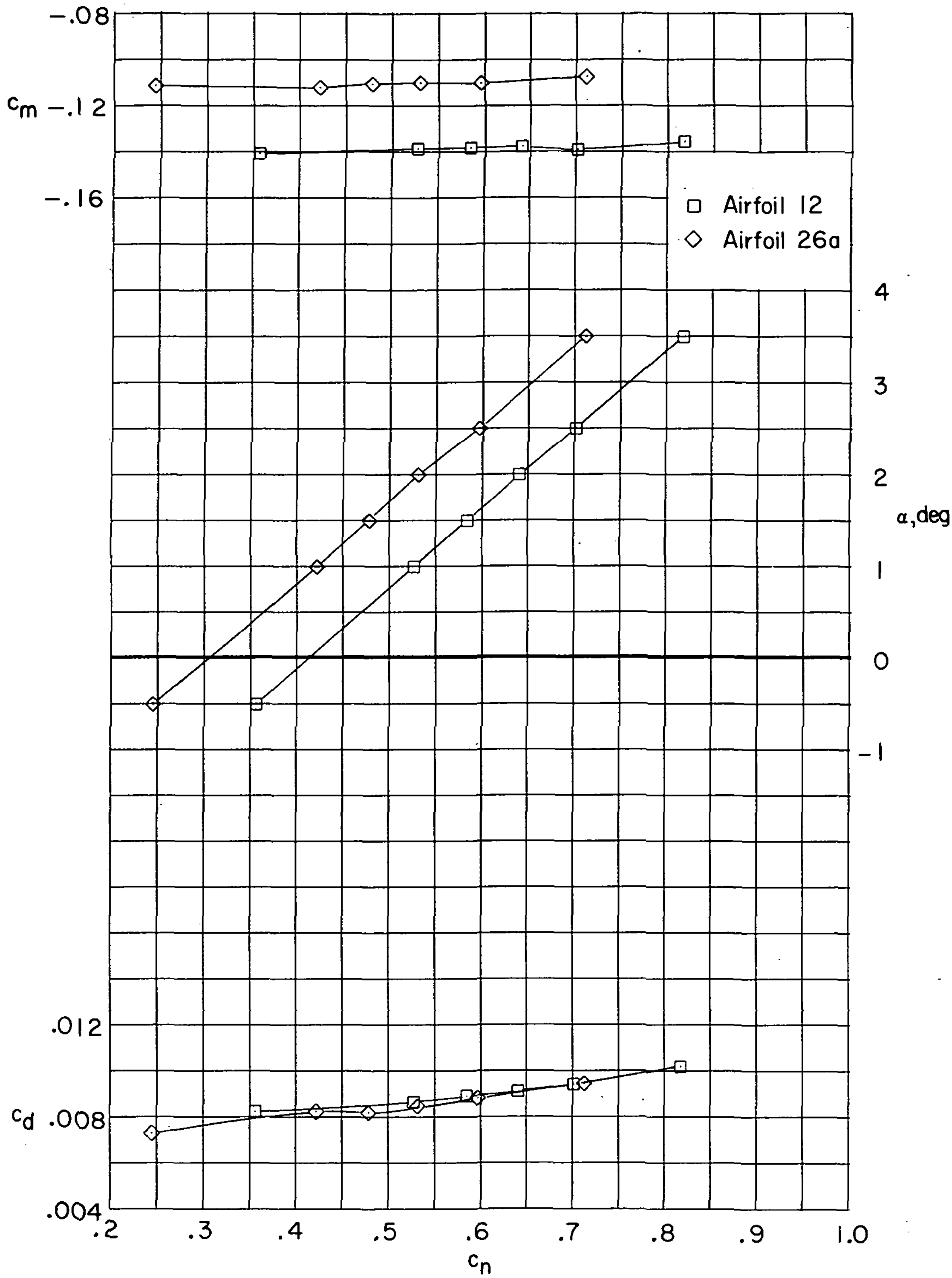


Figure 6.- Variation with Mach number of test wind-tunnel Reynolds number.

[redacted]

[redacted]

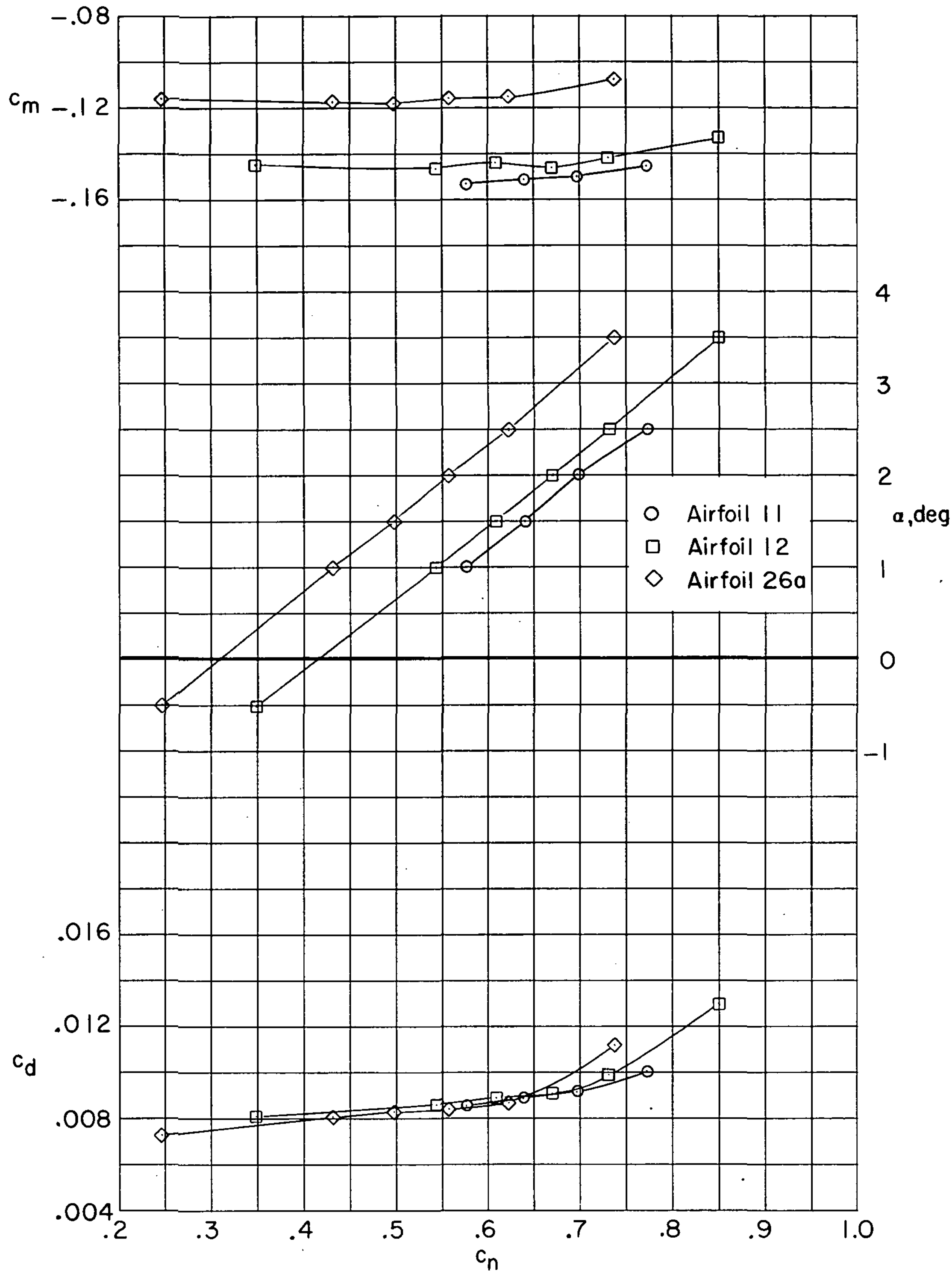


(a) M = 0.50.

Figure 7.- Comparison of force and moment characteristics of supercritical airfoils 11, 12, and 26a.

[redacted]

[redacted]

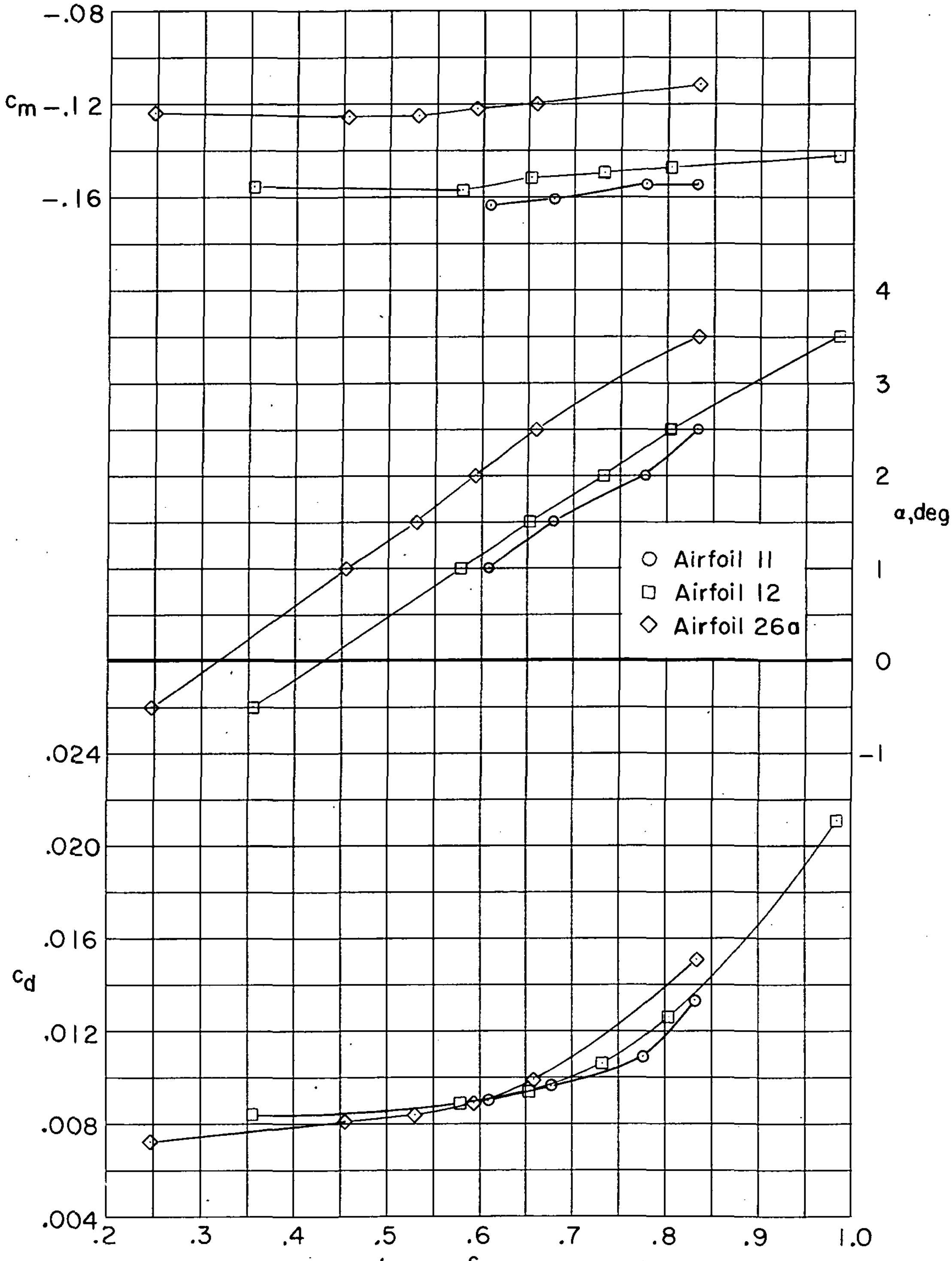


(b) M = 0.60.

Figure 7.- Continued.

[redacted]

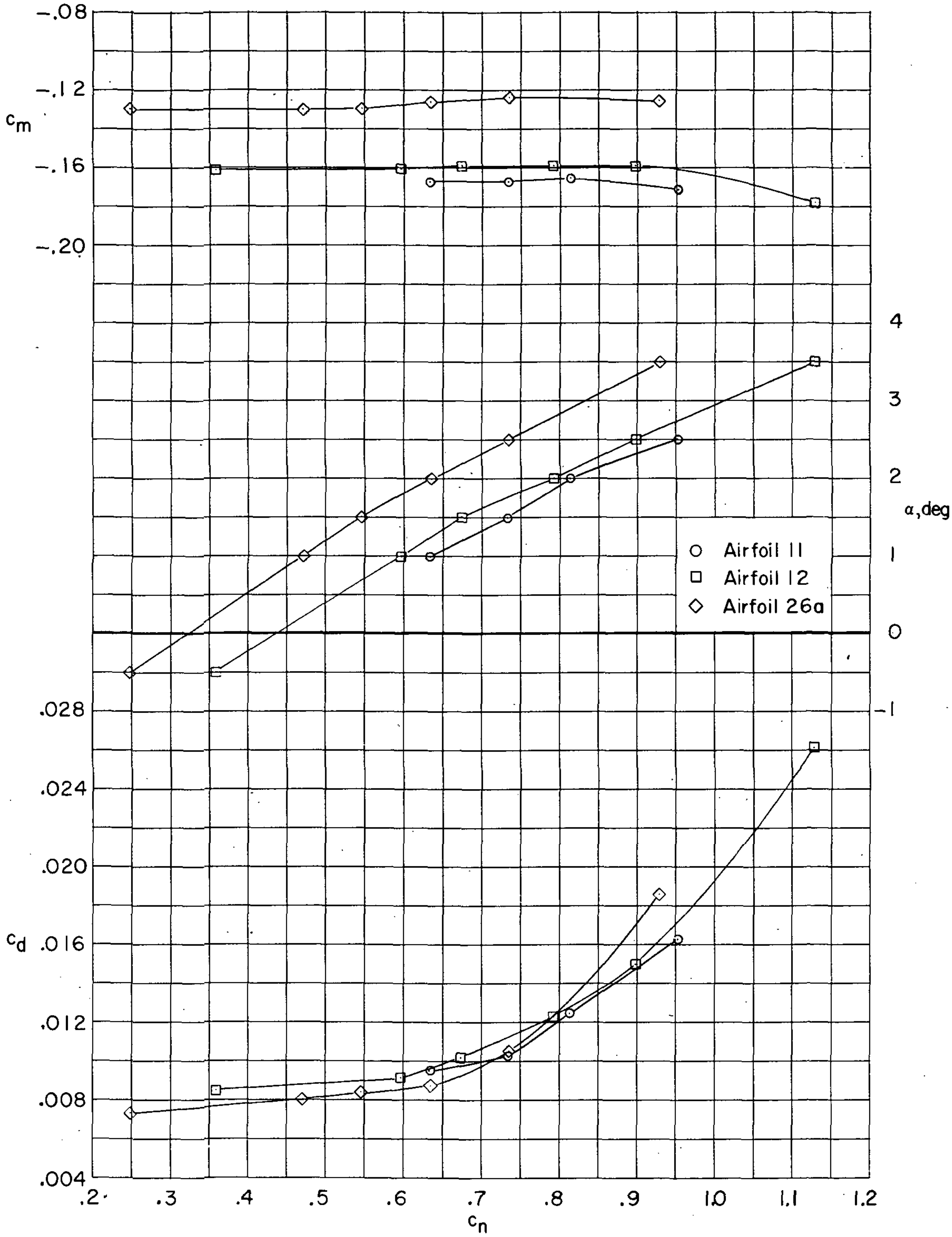
[redacted]



(c) M = 0.70.

Figure 7.- Continued.

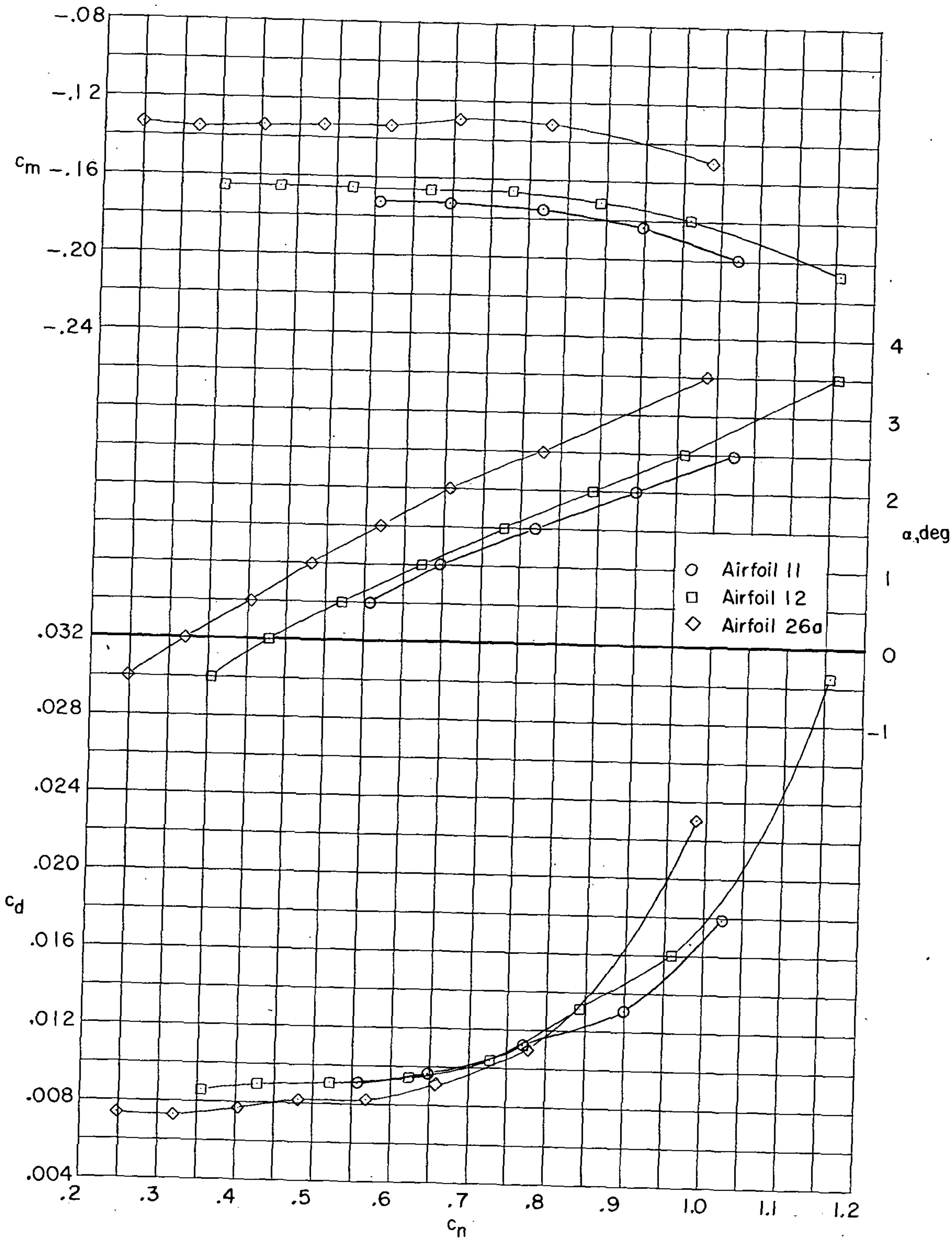
[redacted]



(d) M = 0.74.

Figure 7.- Continued.

[redacted]

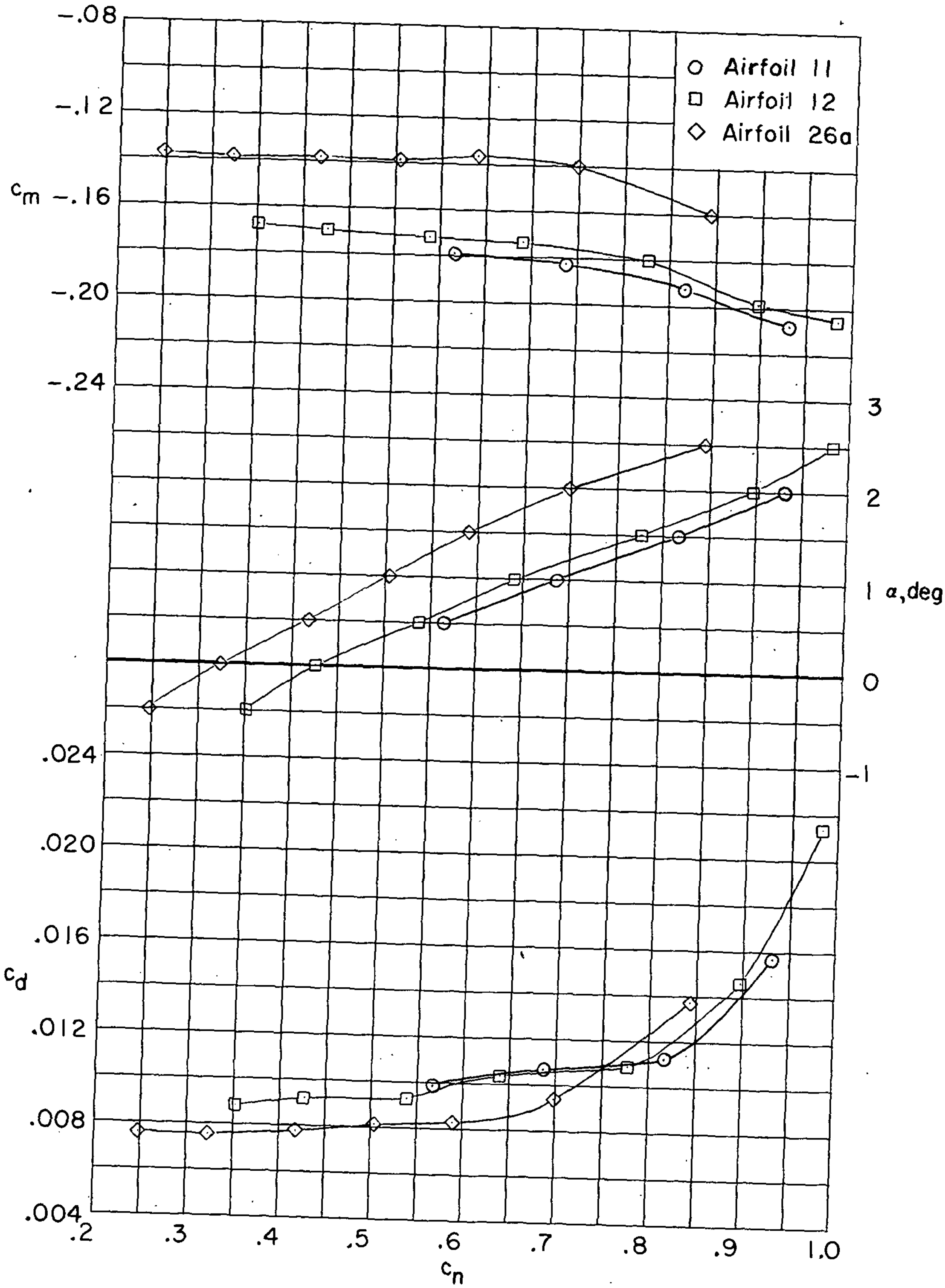


(e) M = 0.76.

Figure 7.- Continued.

[redacted]

[redacted]

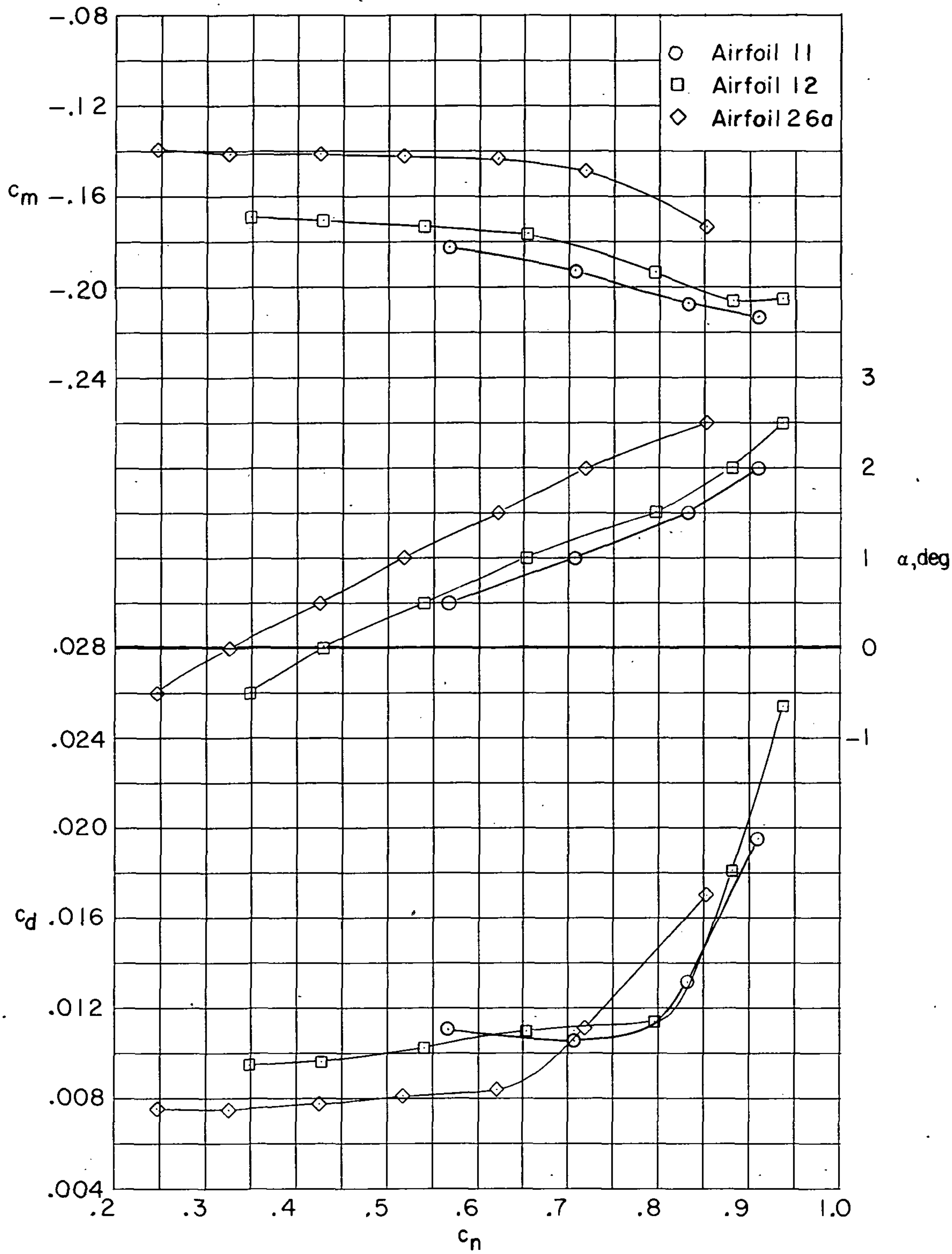


(f) M = 0.78.

Figure 7.- Continued.

[redacted]

[redacted]

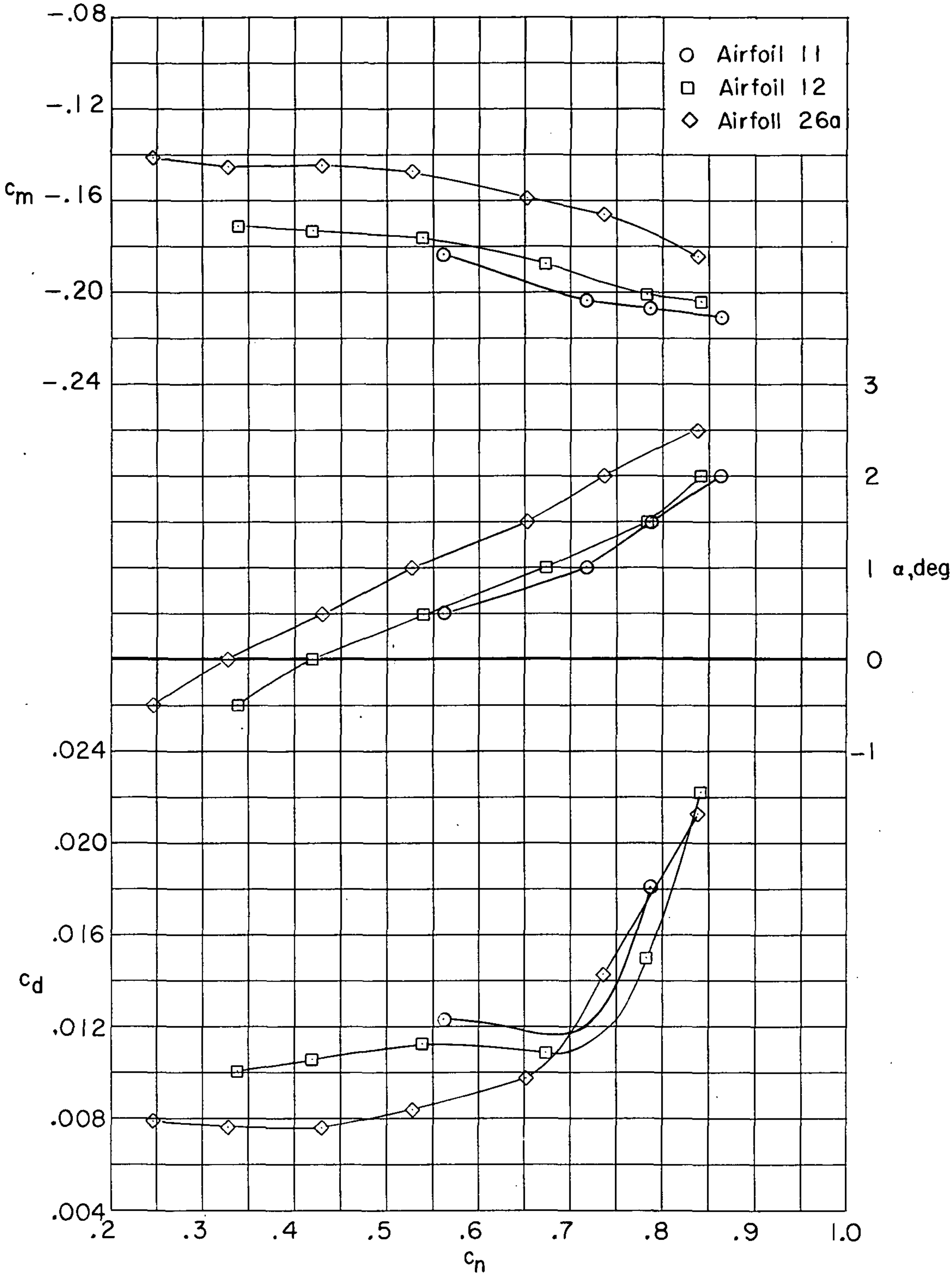


(g) M = 0.79.

Figure 7.- Continued.

[redacted]

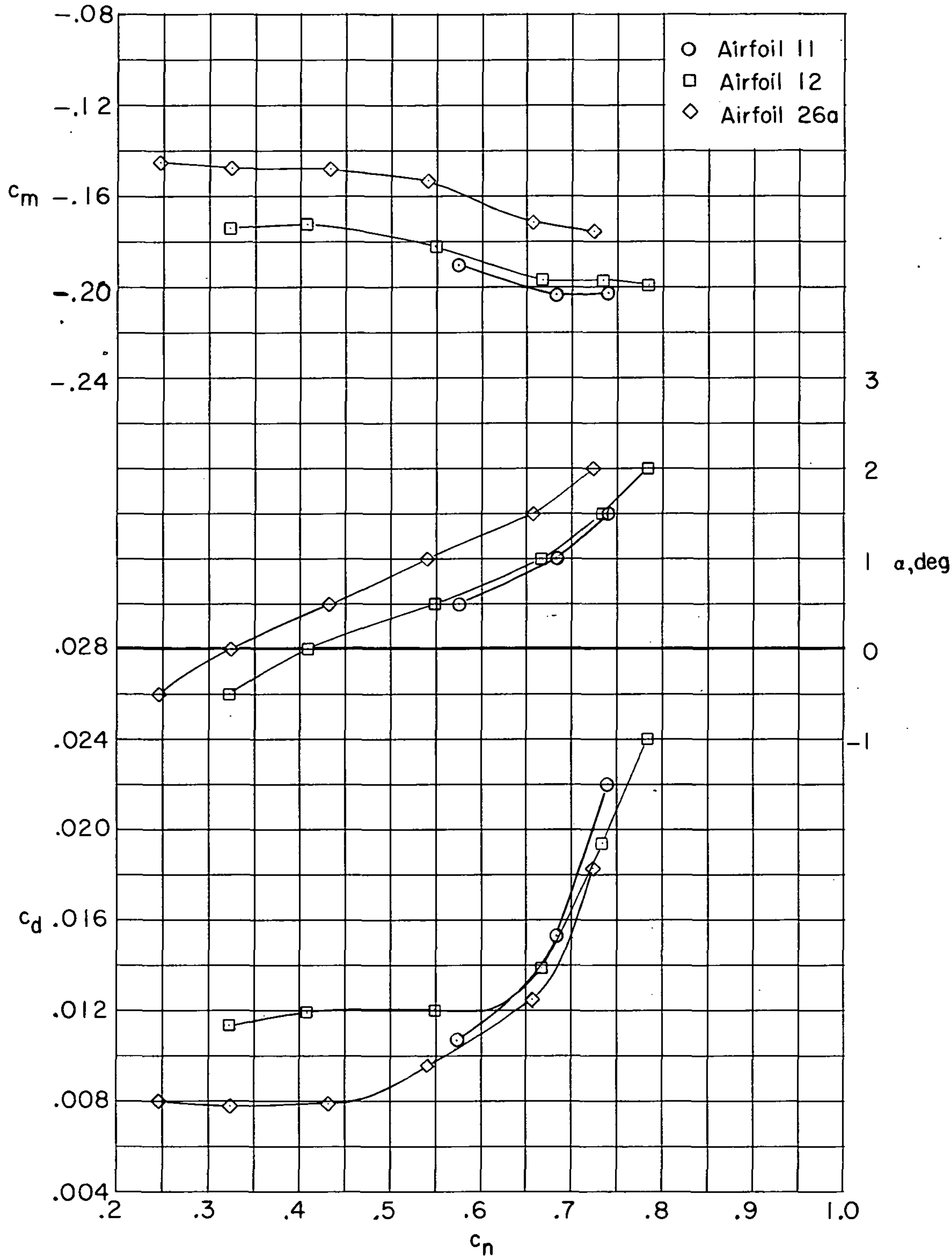
[redacted]



(h) M = 0.80.

Figure 7.- Continued.

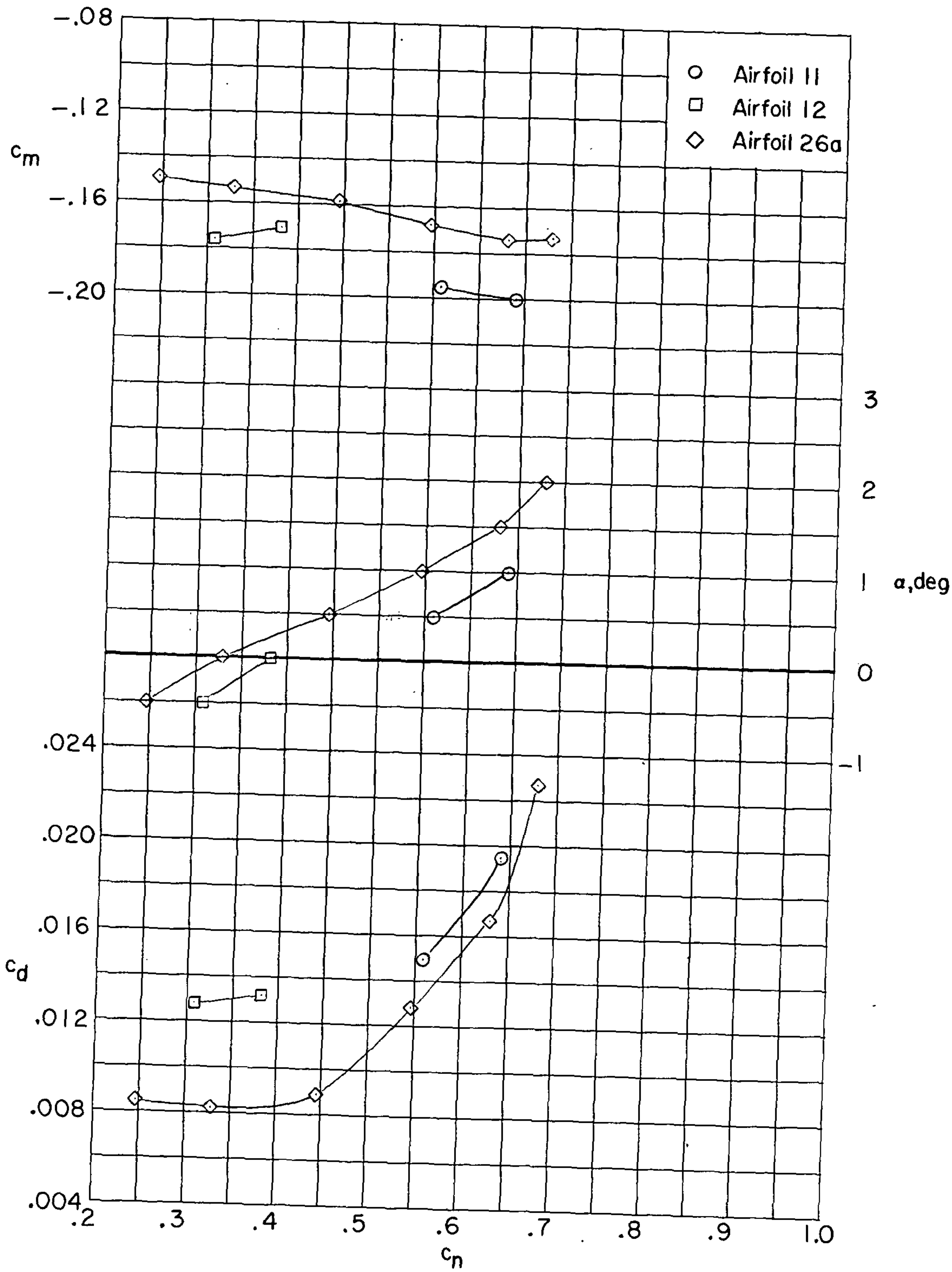
[redacted]



(i) M = 0.81.

Figure 7.- Continued.

[redacted]

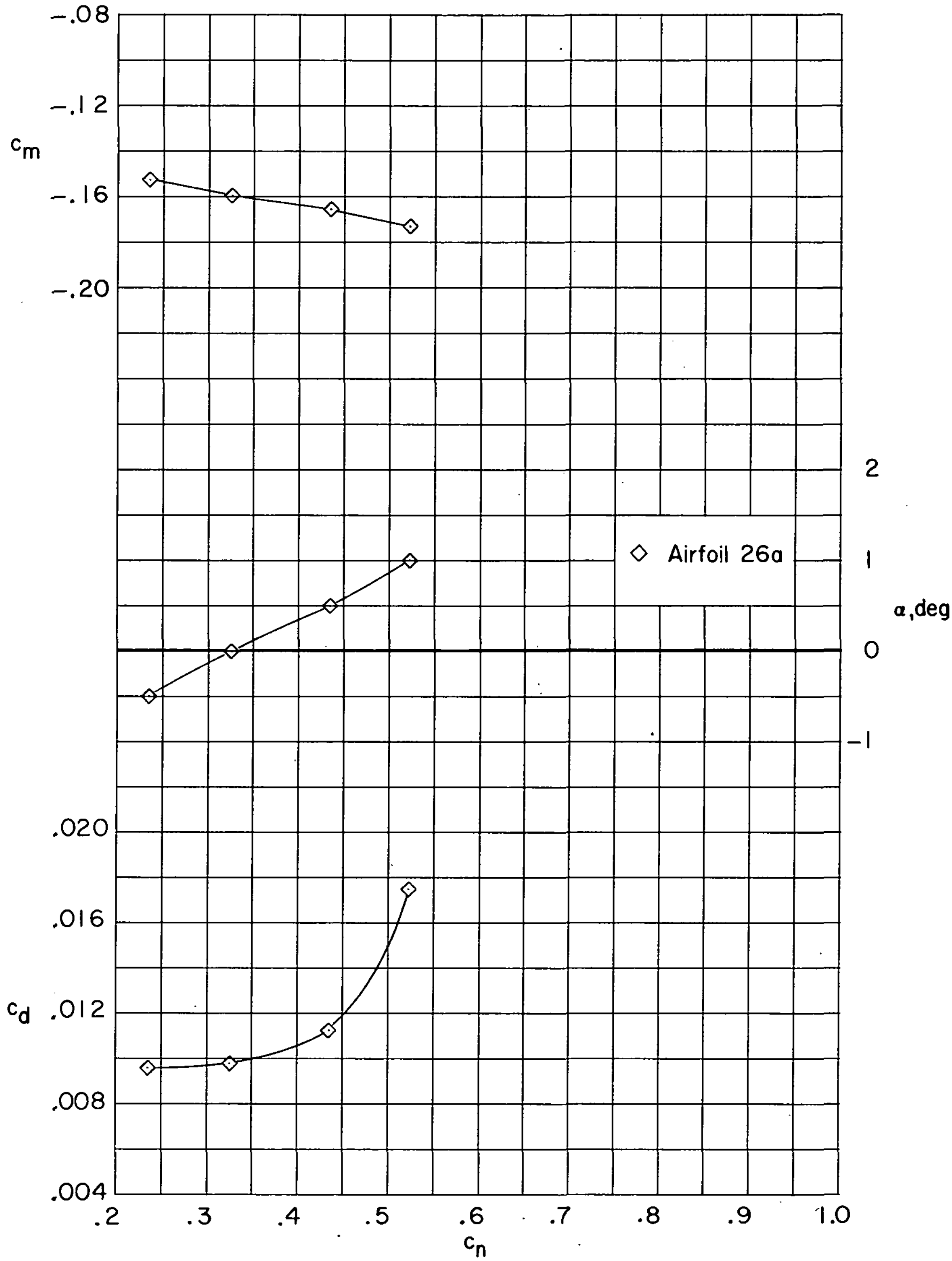


(j) M = 0.82.

Figure 7.- Continued.

[redacted]

[redacted]



(k) M = 0.83.

Figure 7.- Concluded.

[redacted]

[redacted]

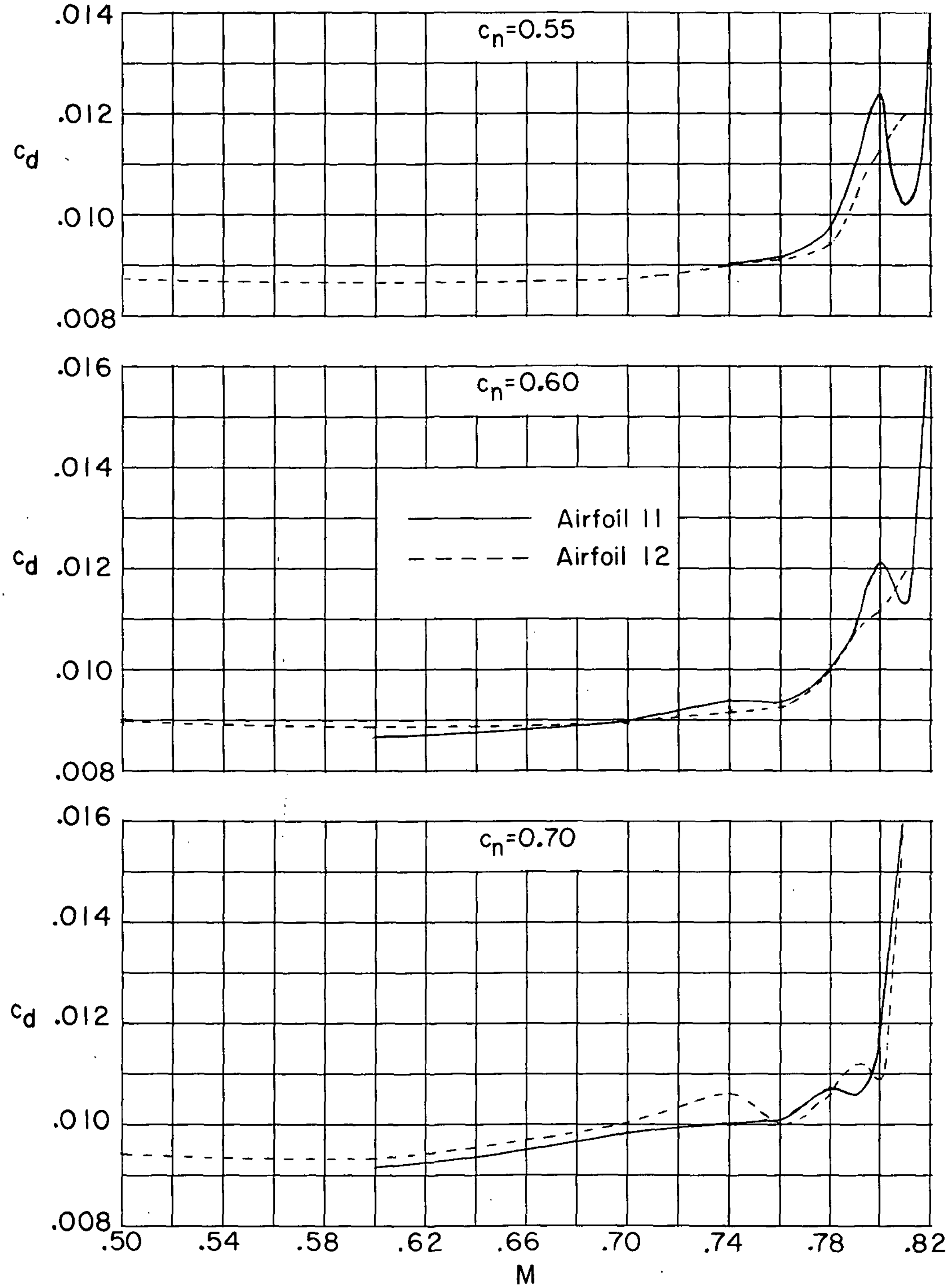


Figure 8.- Variation of section drag coefficient with Mach number of supercritical airfoils 11 and 12 at various normal-force coefficients.

[redacted]

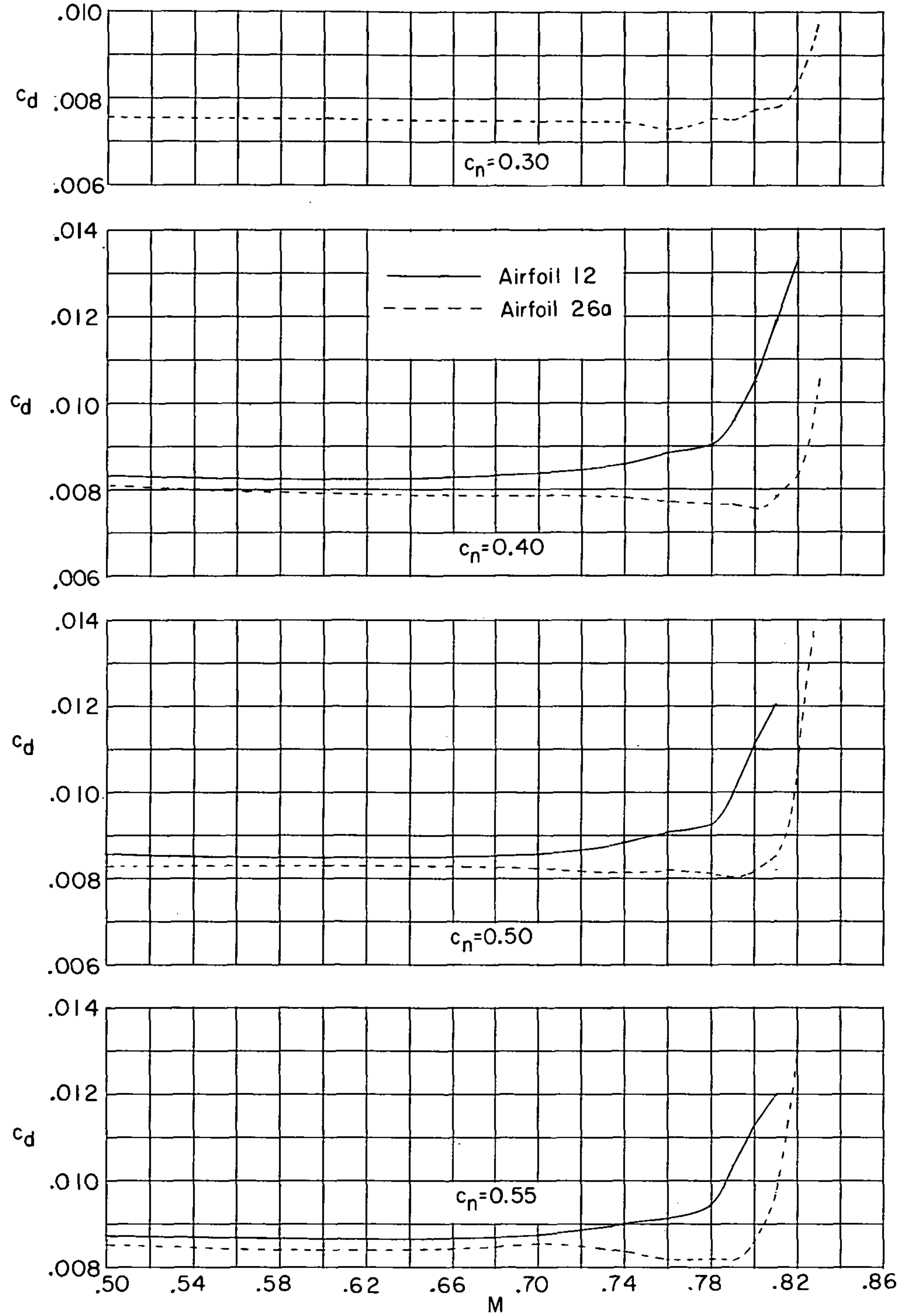


Figure 9.- Variation of section drag coefficient with Mach number of supercritical airfoils 12 and 26a at various normal-force coefficients.

[redacted]

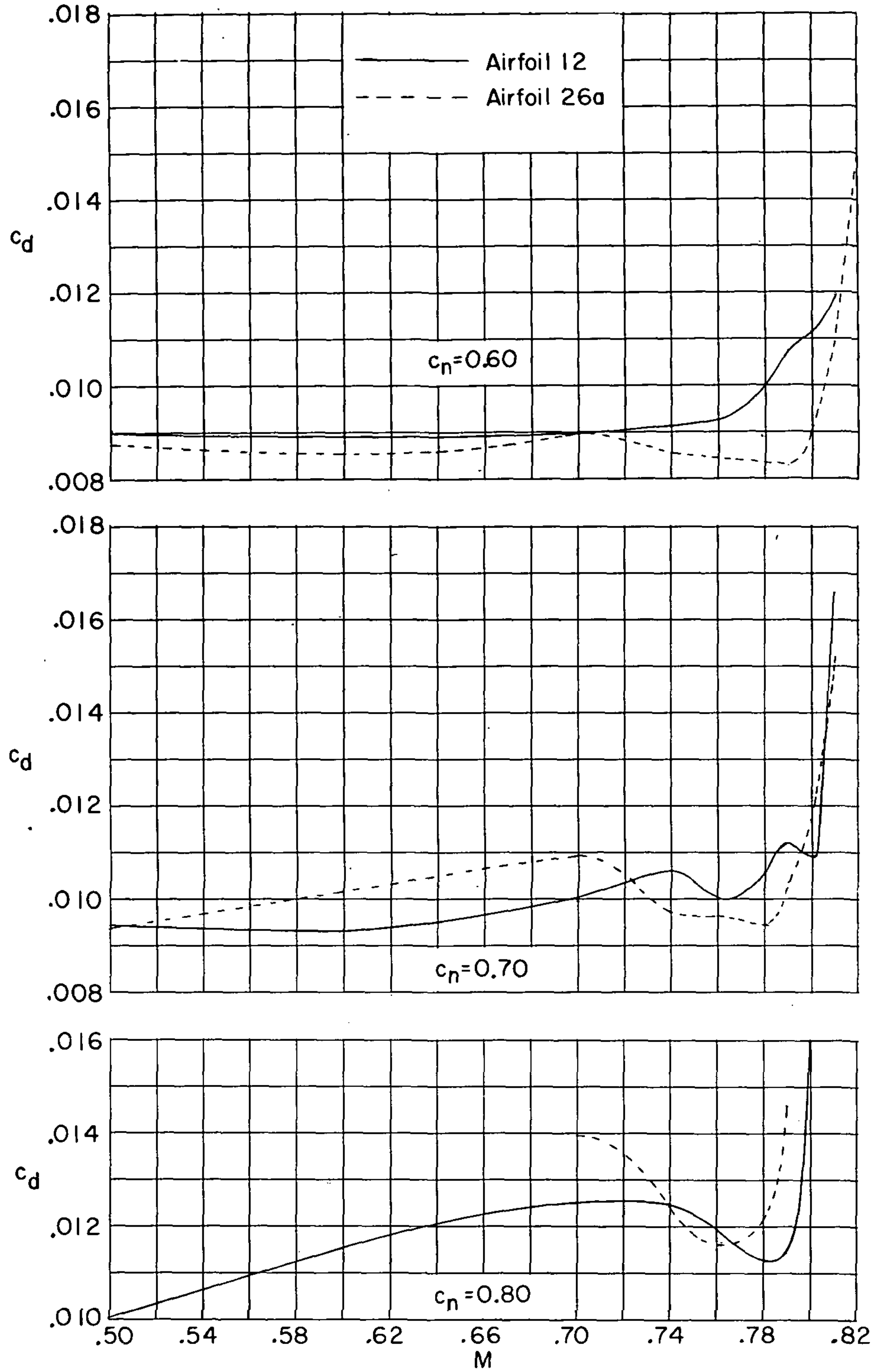


Figure 9.- Concluded.

[redacted]

[redacted]

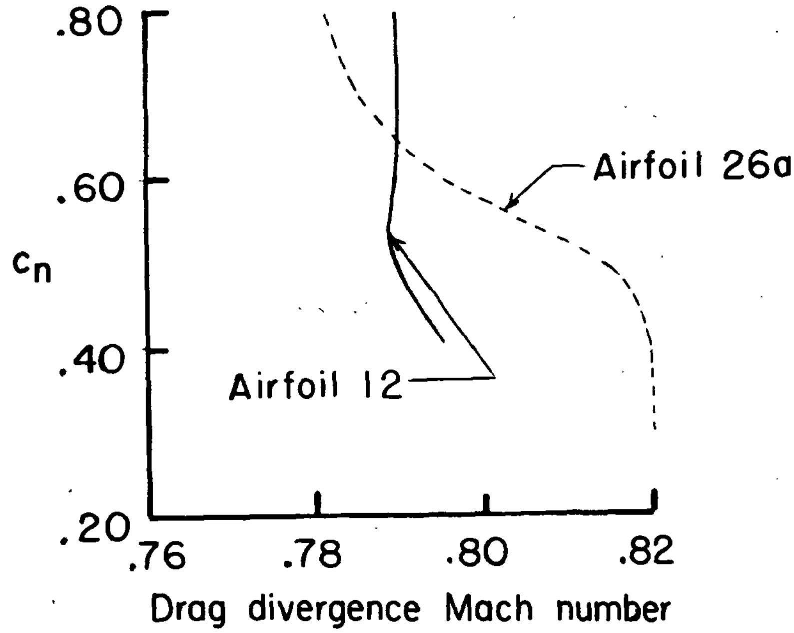


Figure 10.- Drag-divergence Mach numbers.

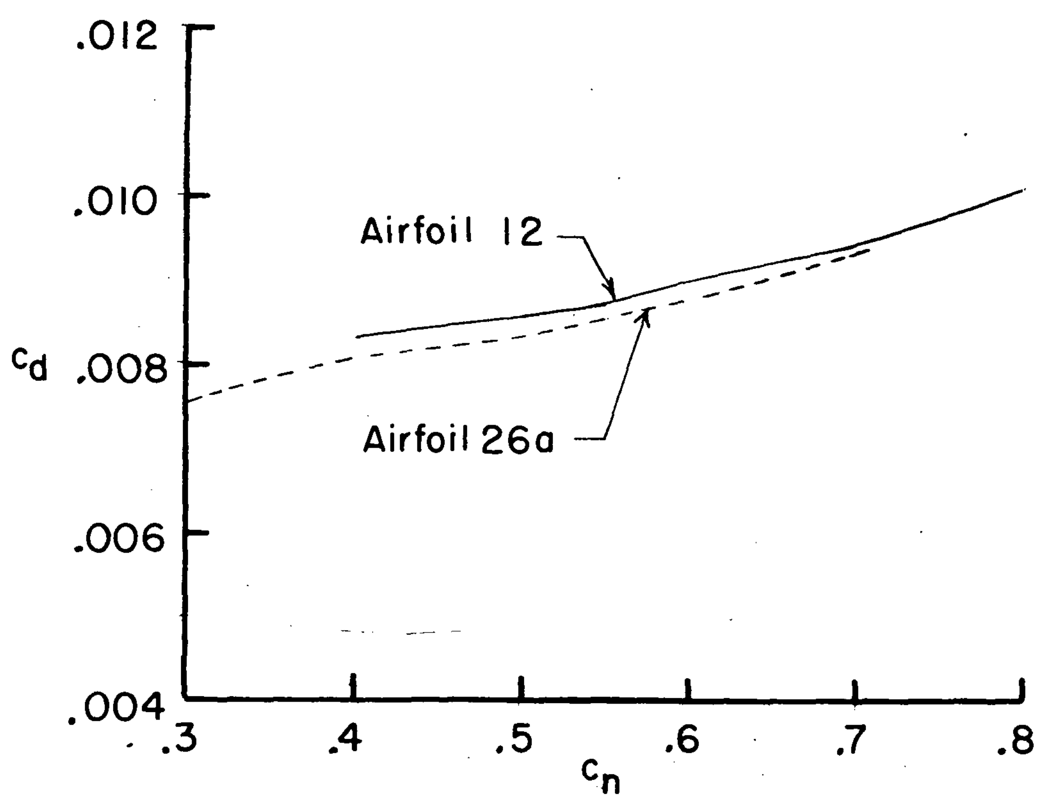
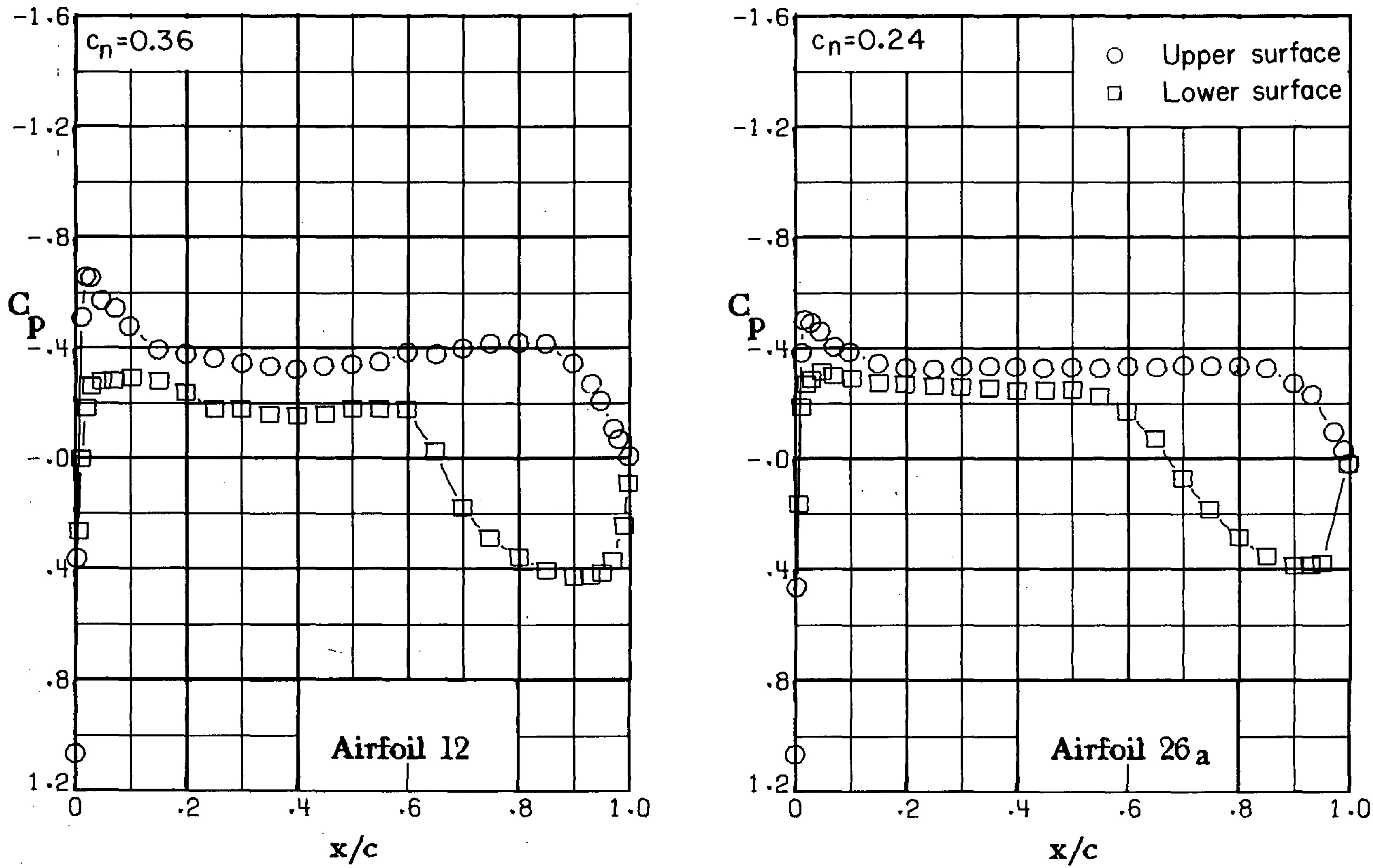


Figure 11.- Subcritical drag levels.

[redacted]

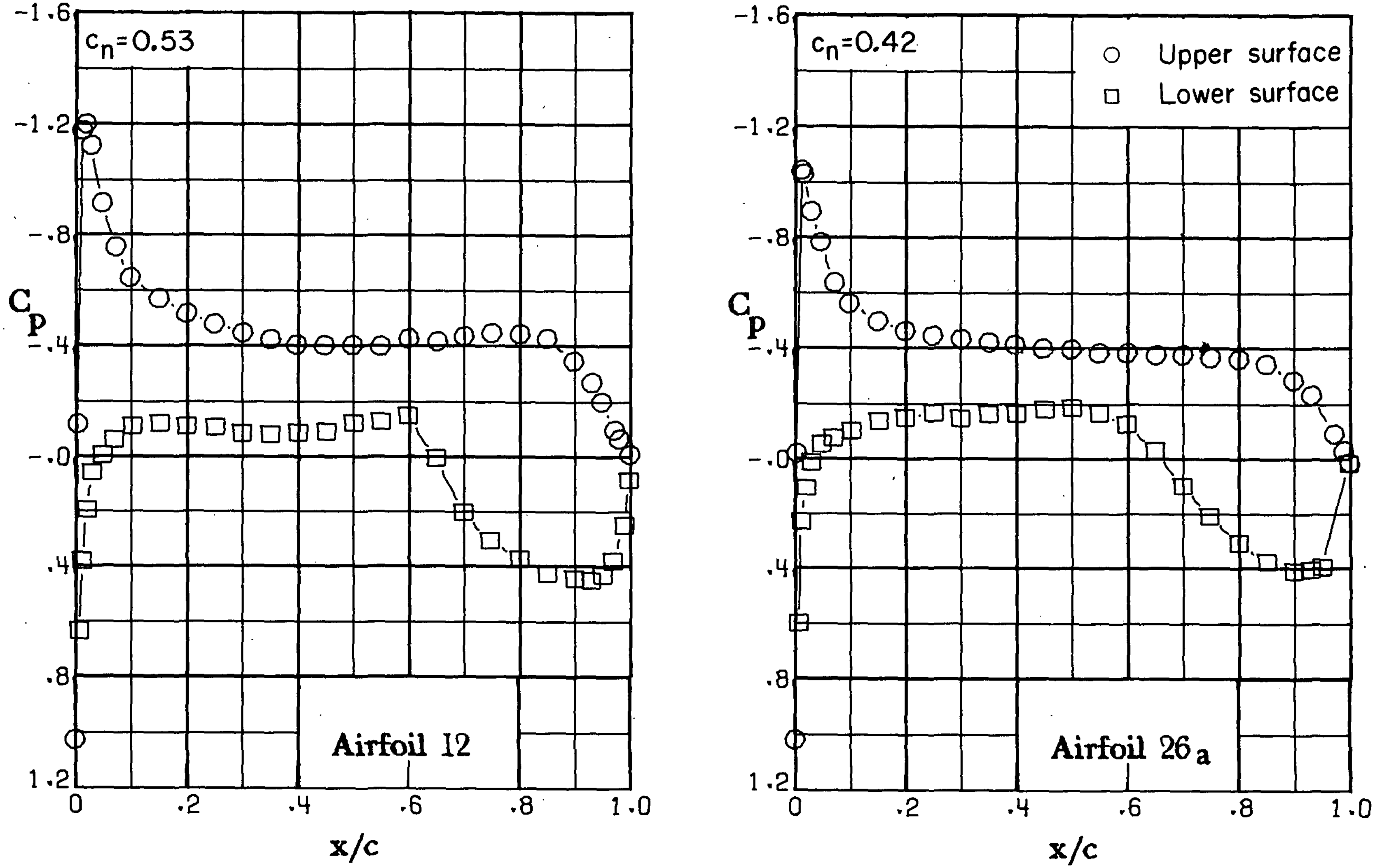
[redacted]



(a) M = 0.50; α = -0.5°.

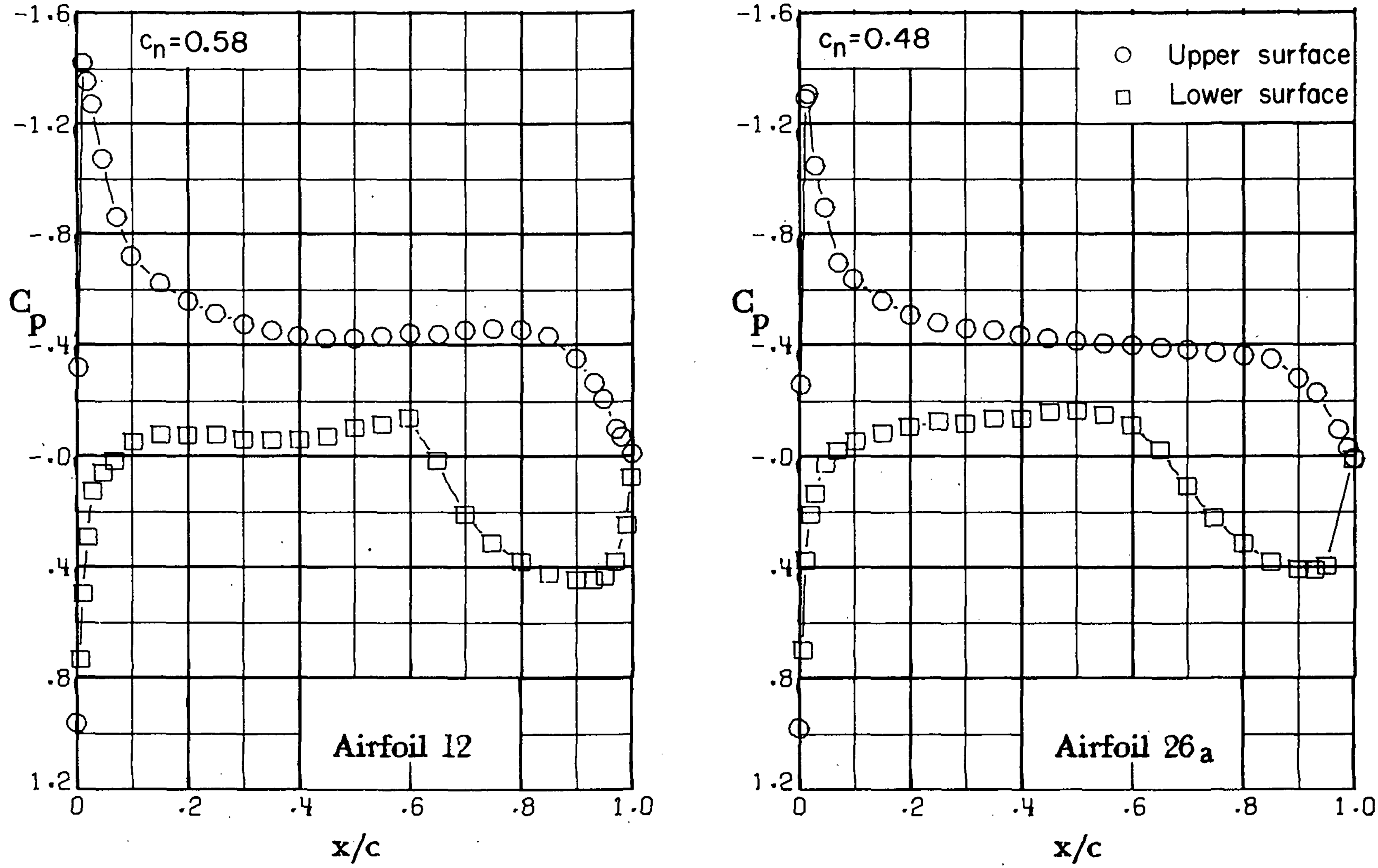
Figure 12.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.50.

[redacted]



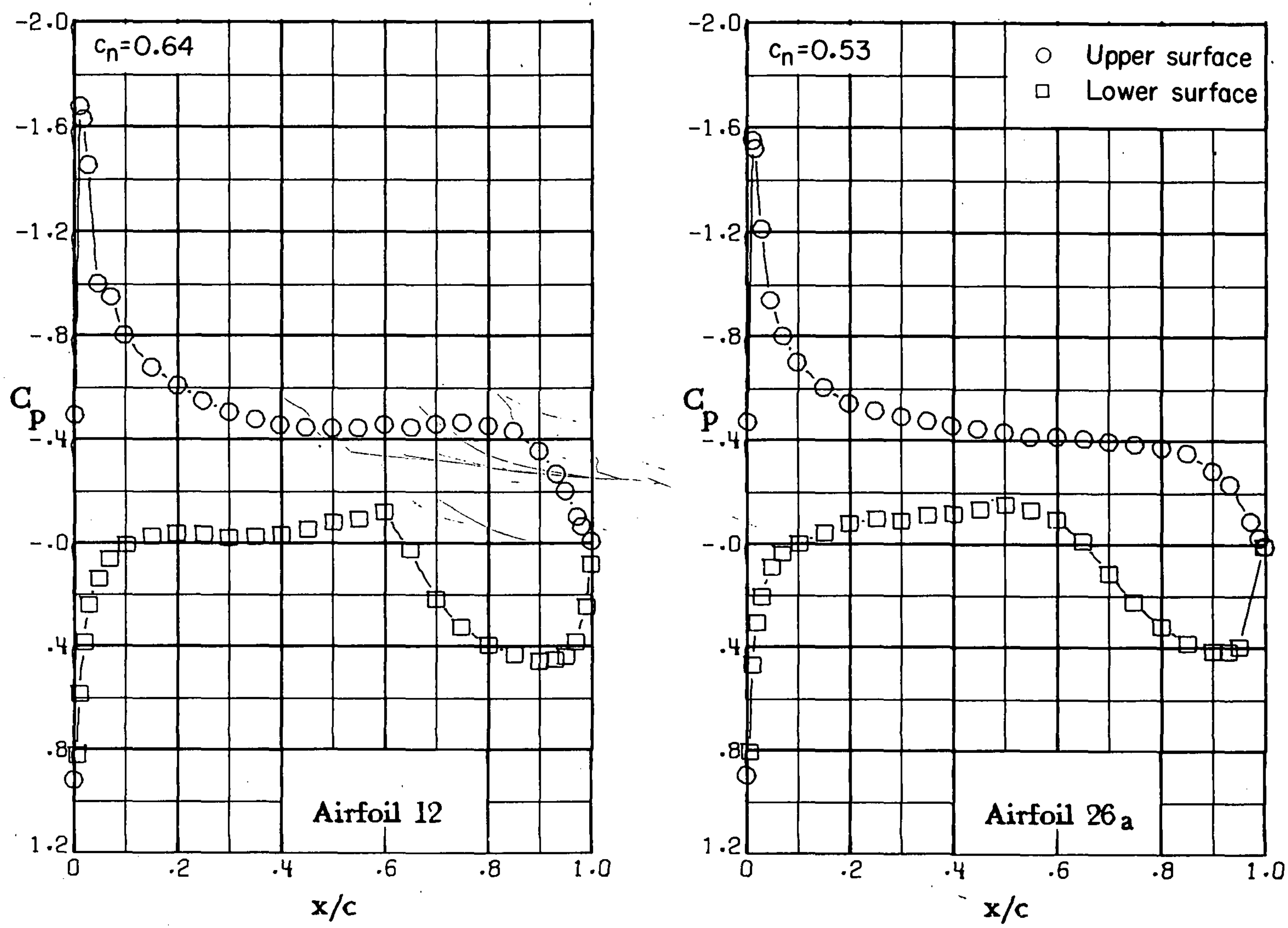
(b) M = 0.50; α = 1.0°.

Figure 12.- Continued.



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

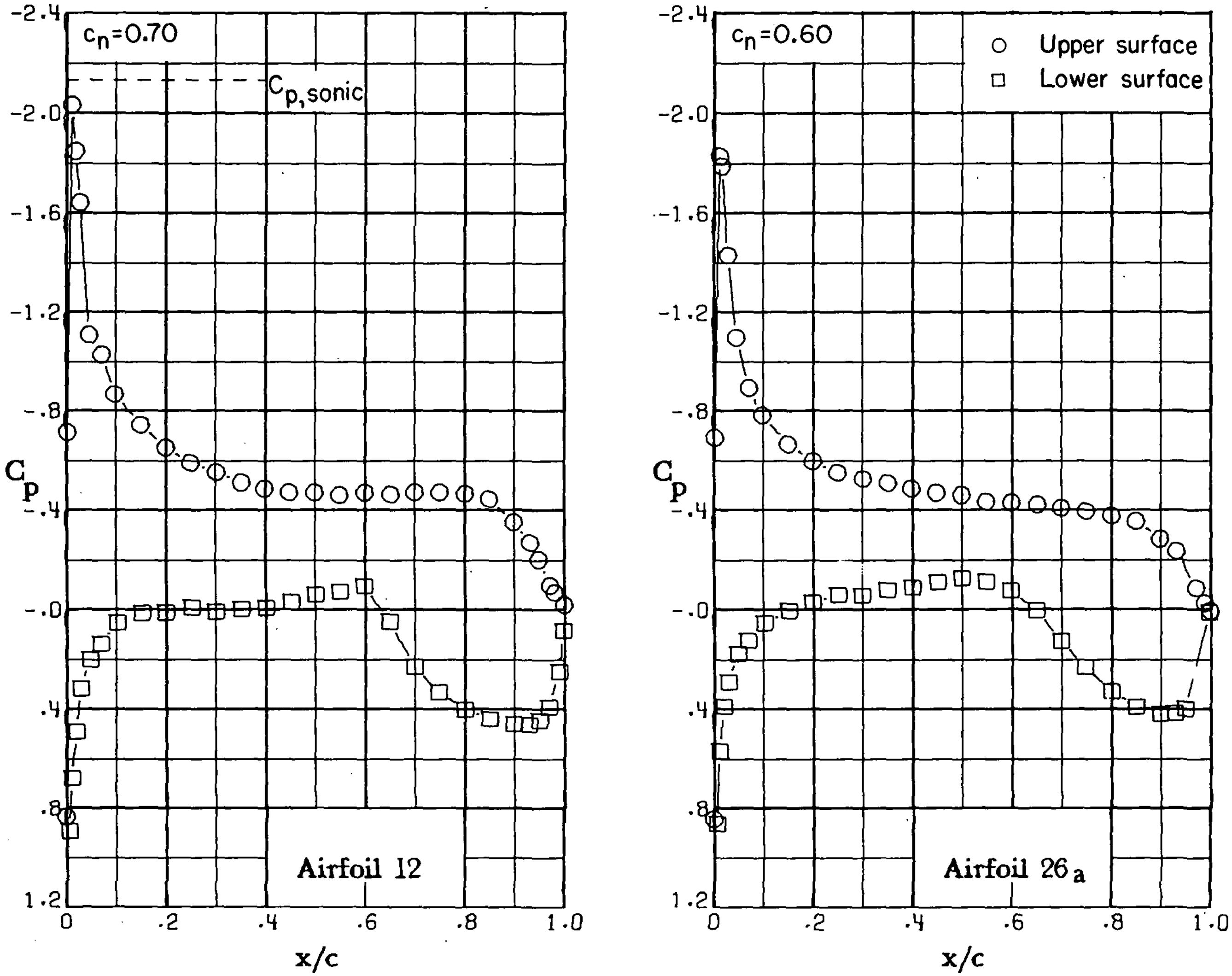
Figure 12.- Continued.



(d) M = 0.50; α = 2.0°.

Figure 12.- Continued.

[redacted]

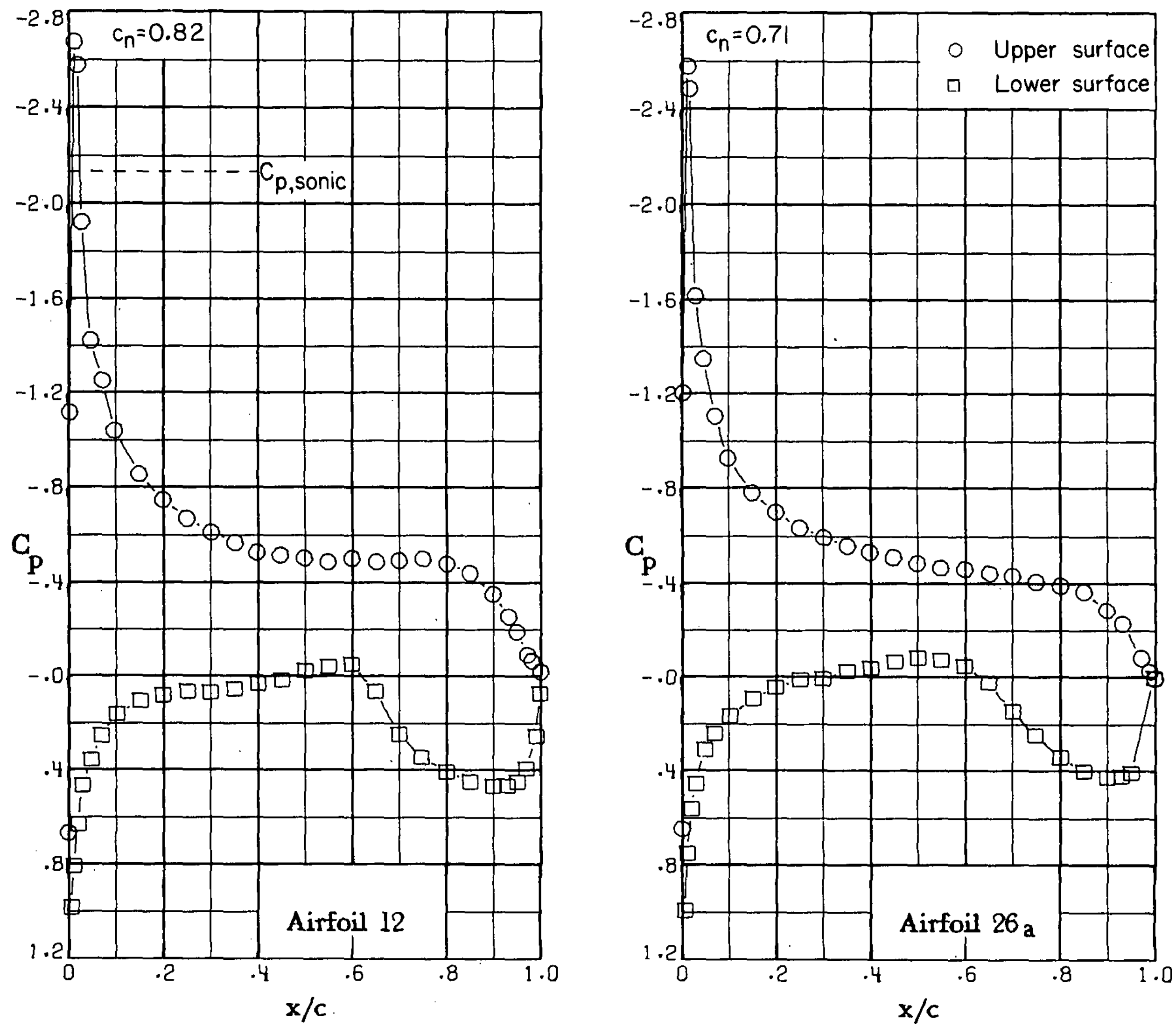


(e) $M = 0.50$; $\alpha = 2.5^{\circ}$.

Figure 12.- Continued.

[redacted]

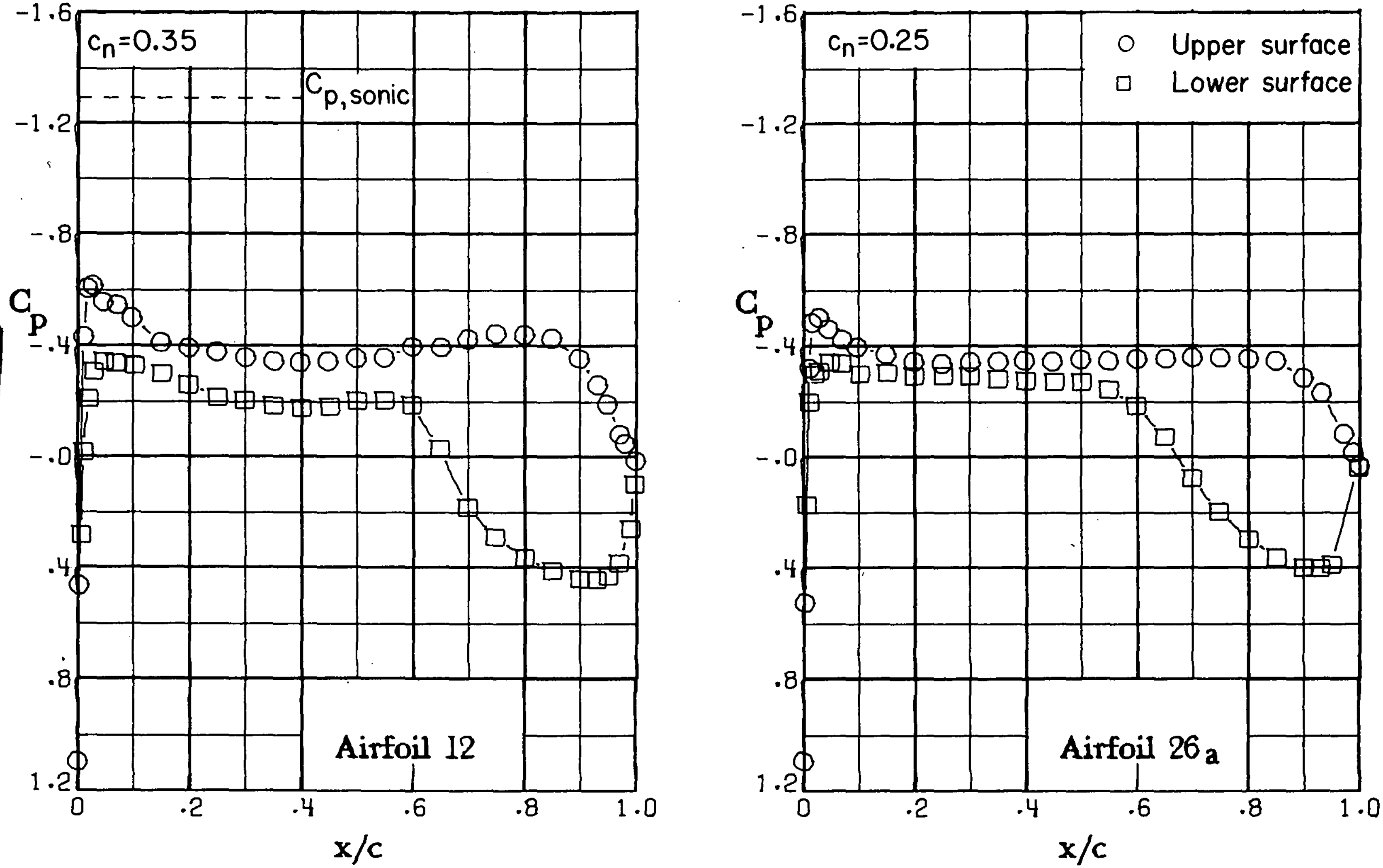
[redacted]



(f) M = 0.50; α = 3.5°.

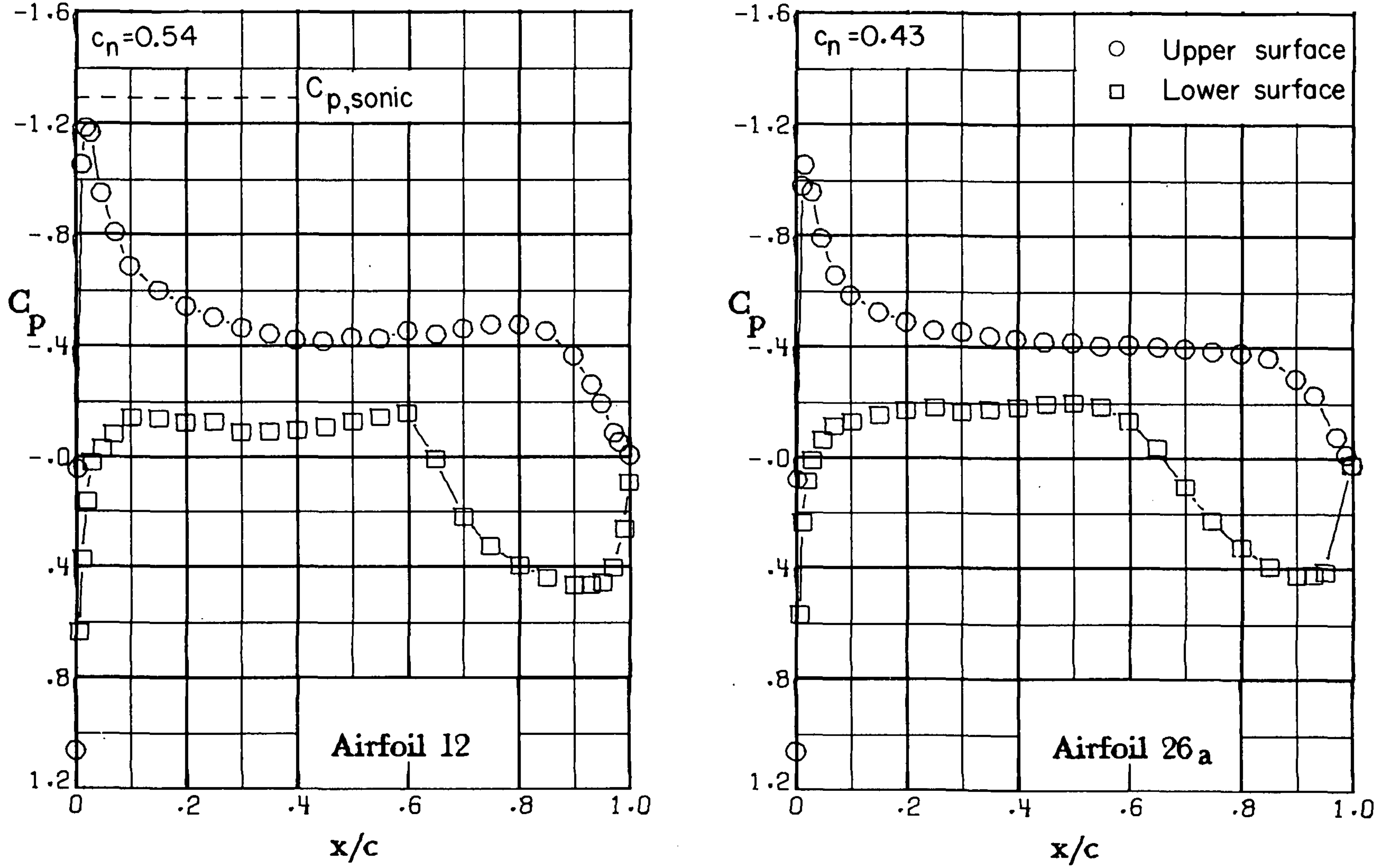
Figure 12.- Concluded.

[redacted]



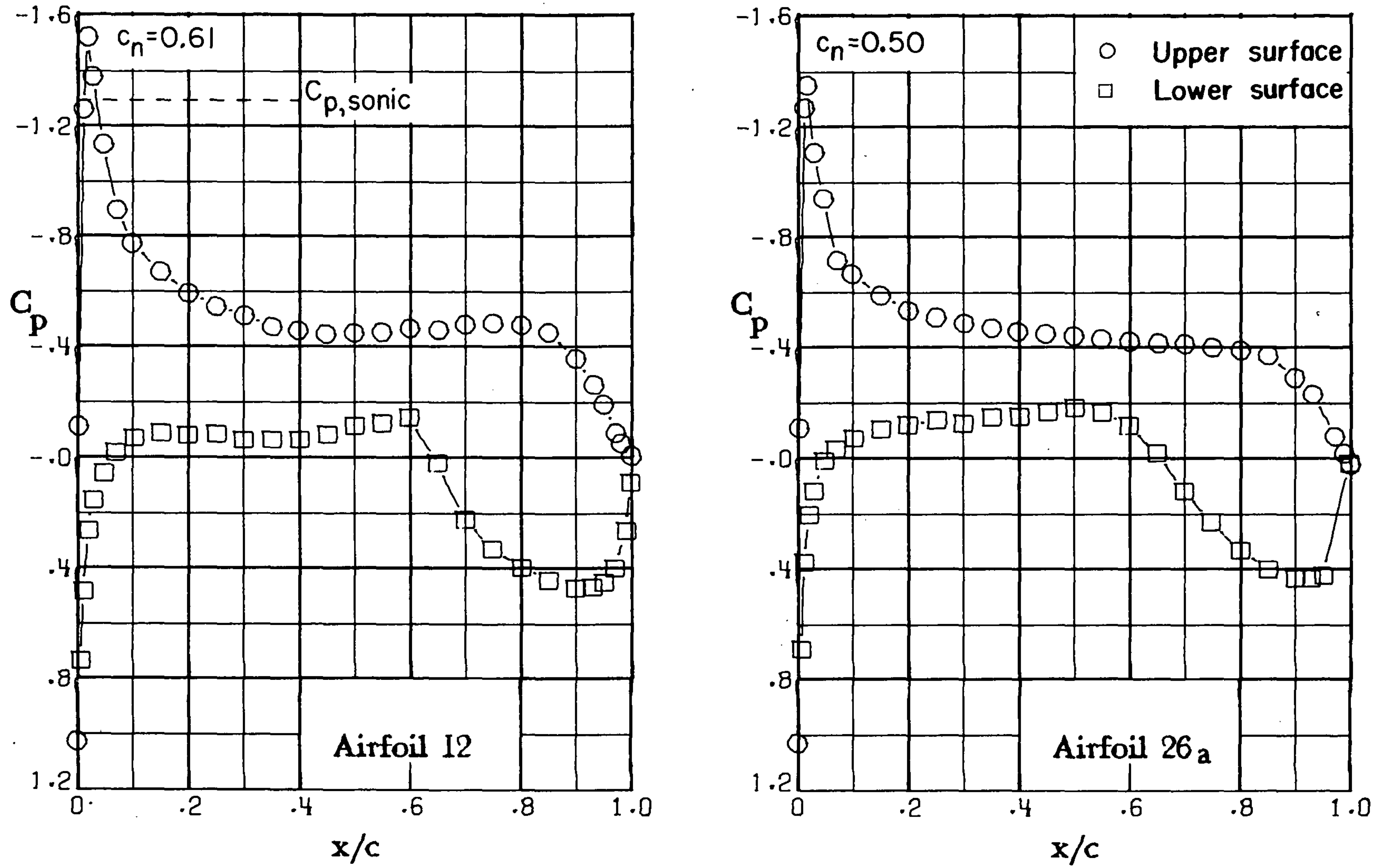
(a) M = 0.60; α = -0.5°.

Figure 13.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.60.



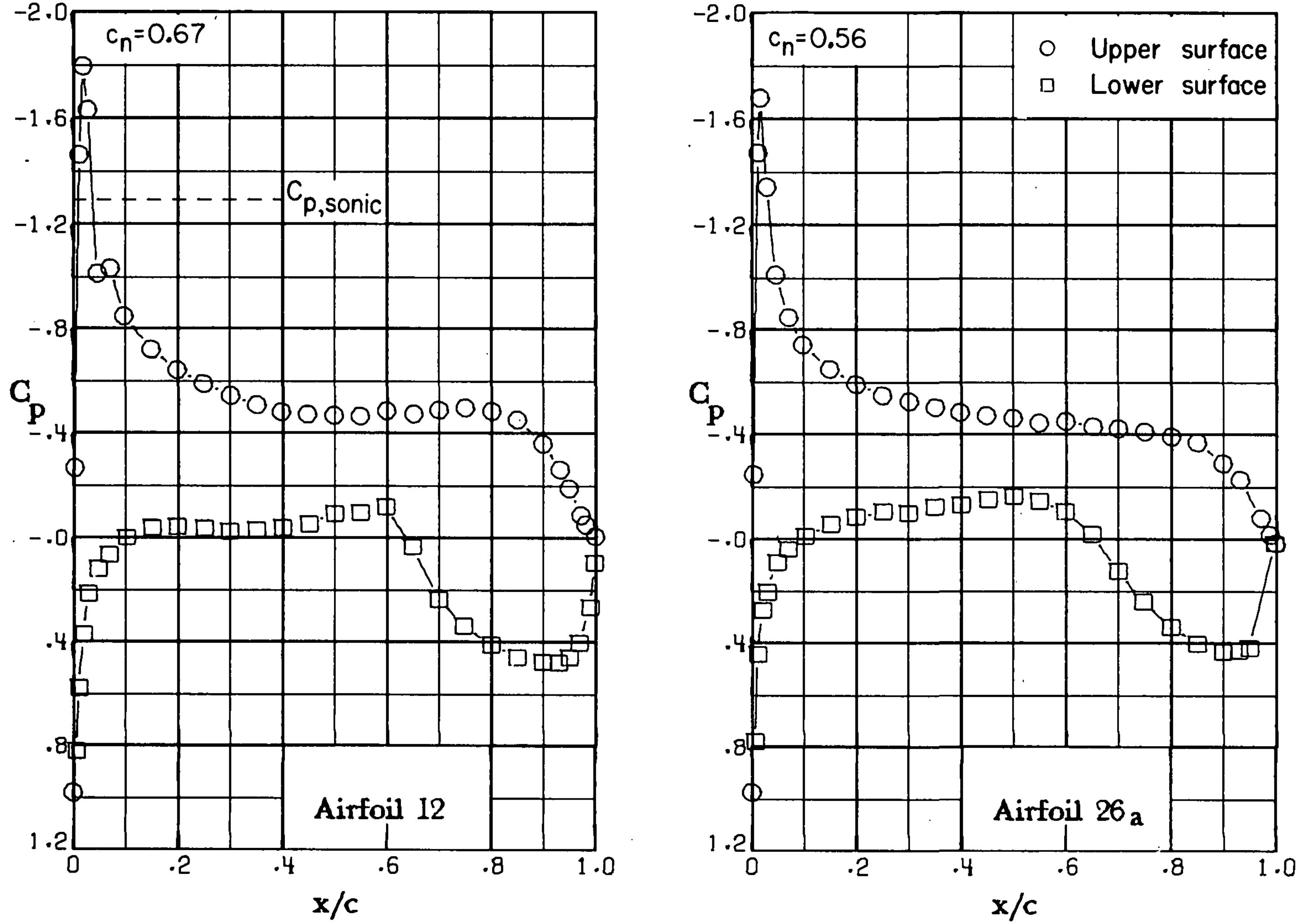
(b) M = 0.60; α = 1.0°.

Figure 13.- Continued.



(c) M = 0.60; α = 1.5°.

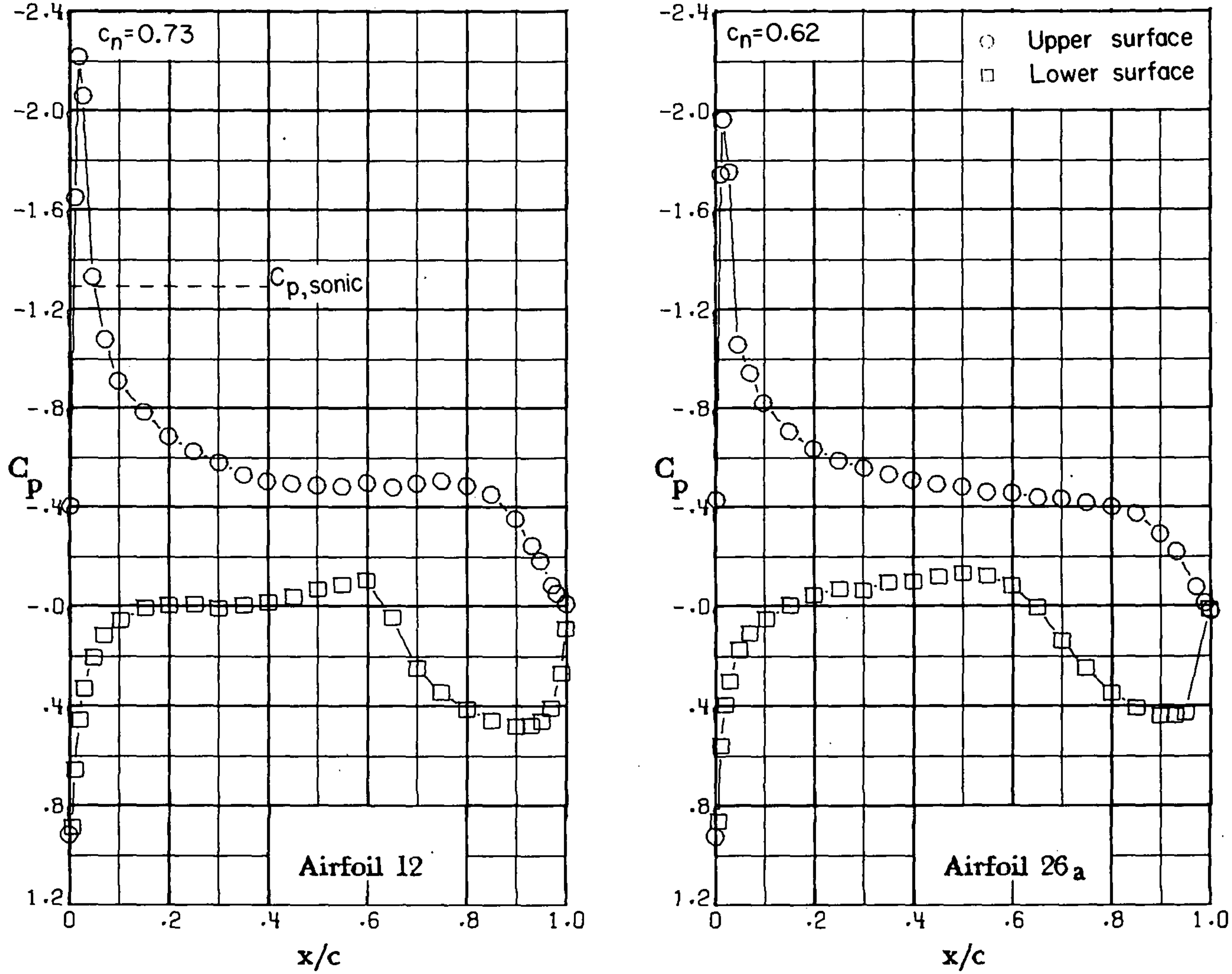
Figure 13.- Continued.



(d) M = 0.60; α = 2.0°.

Figure 13.- Continued.

[redacted]

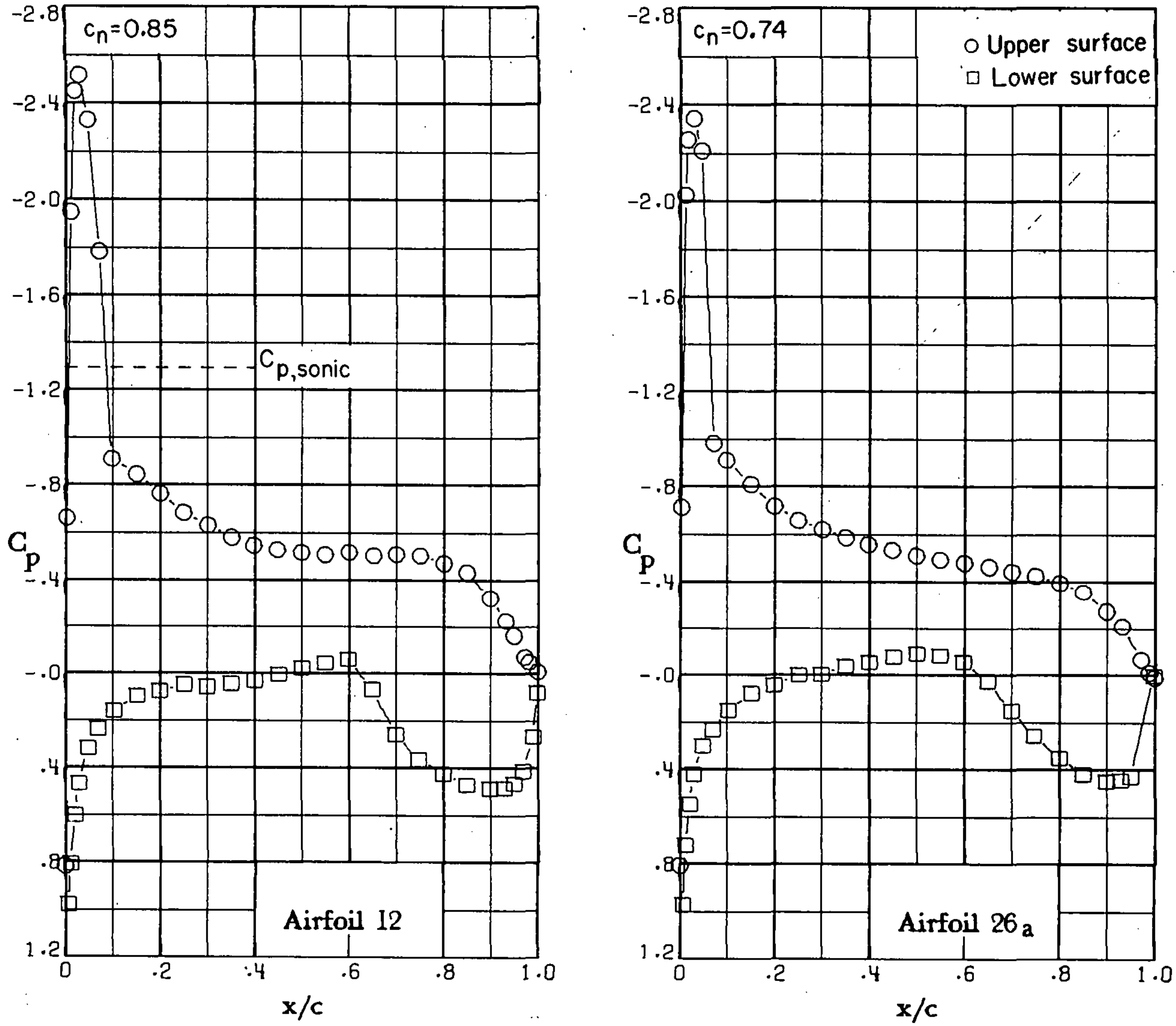


(e) $M = 0.60$; $\alpha = 2.5^o$.

Figure 13.- Continued.

[redacted]

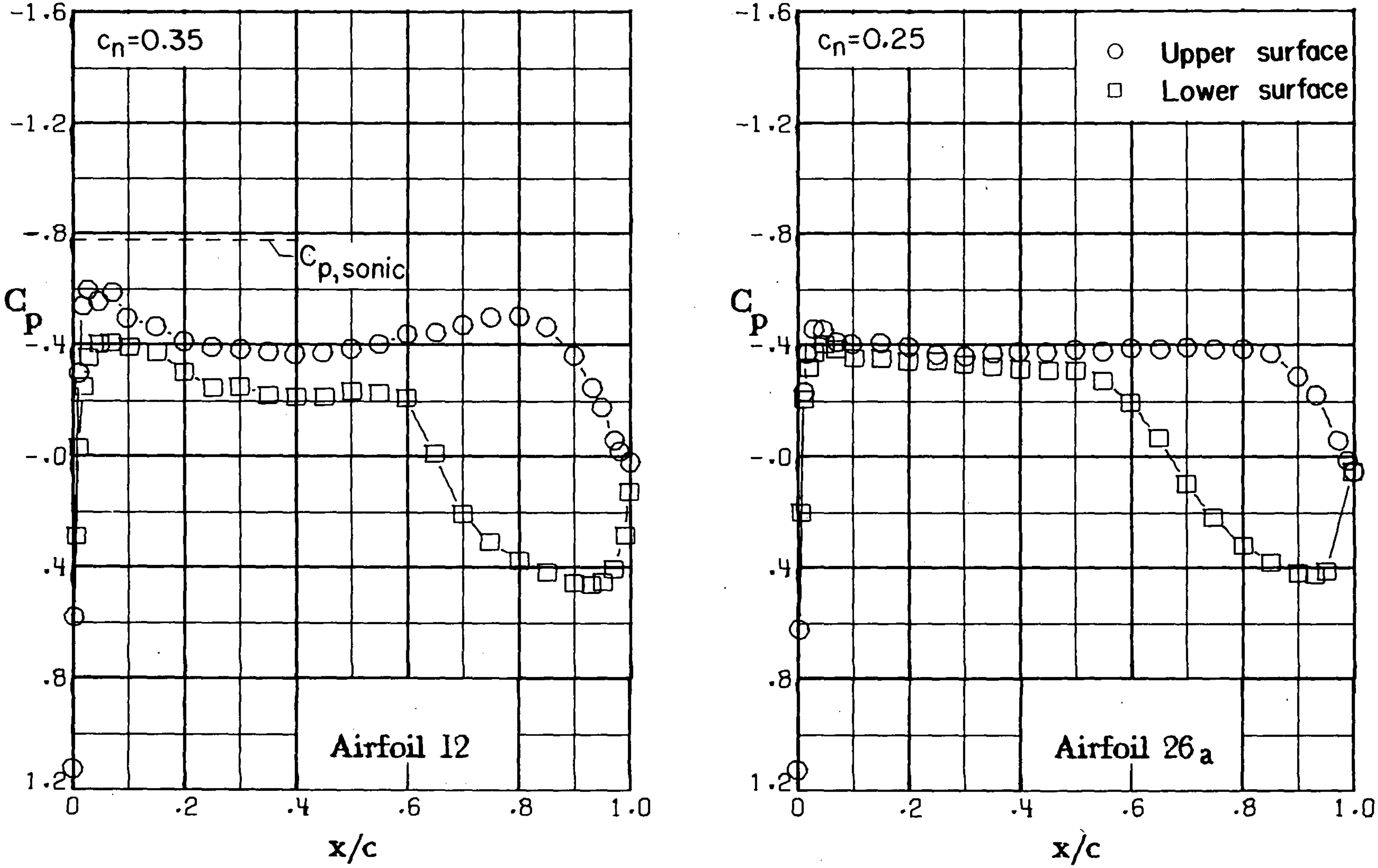
[redacted]



(f) M = 0.60; α = 3.5°.

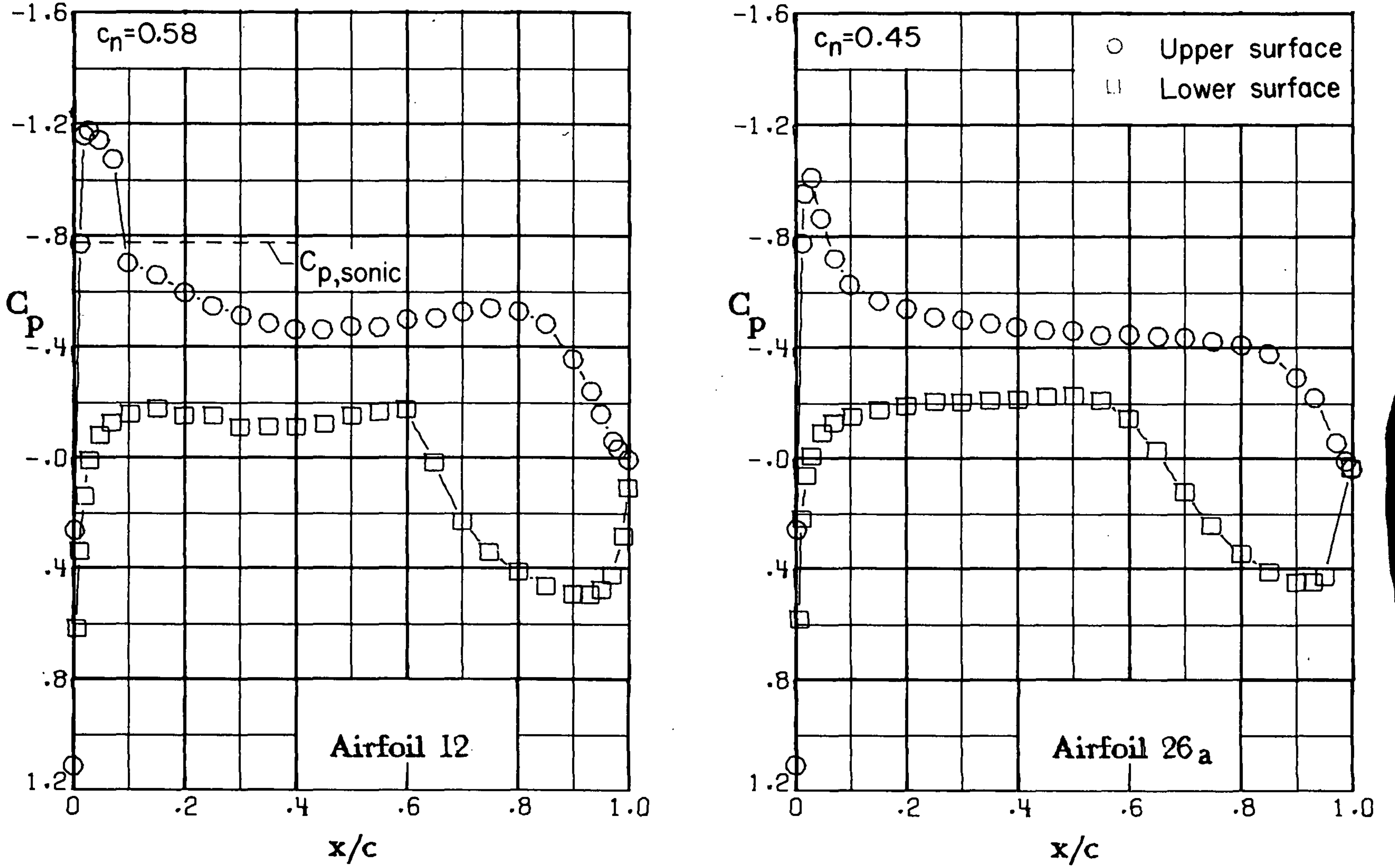
Figure 13.- Concluded.

[redacted]



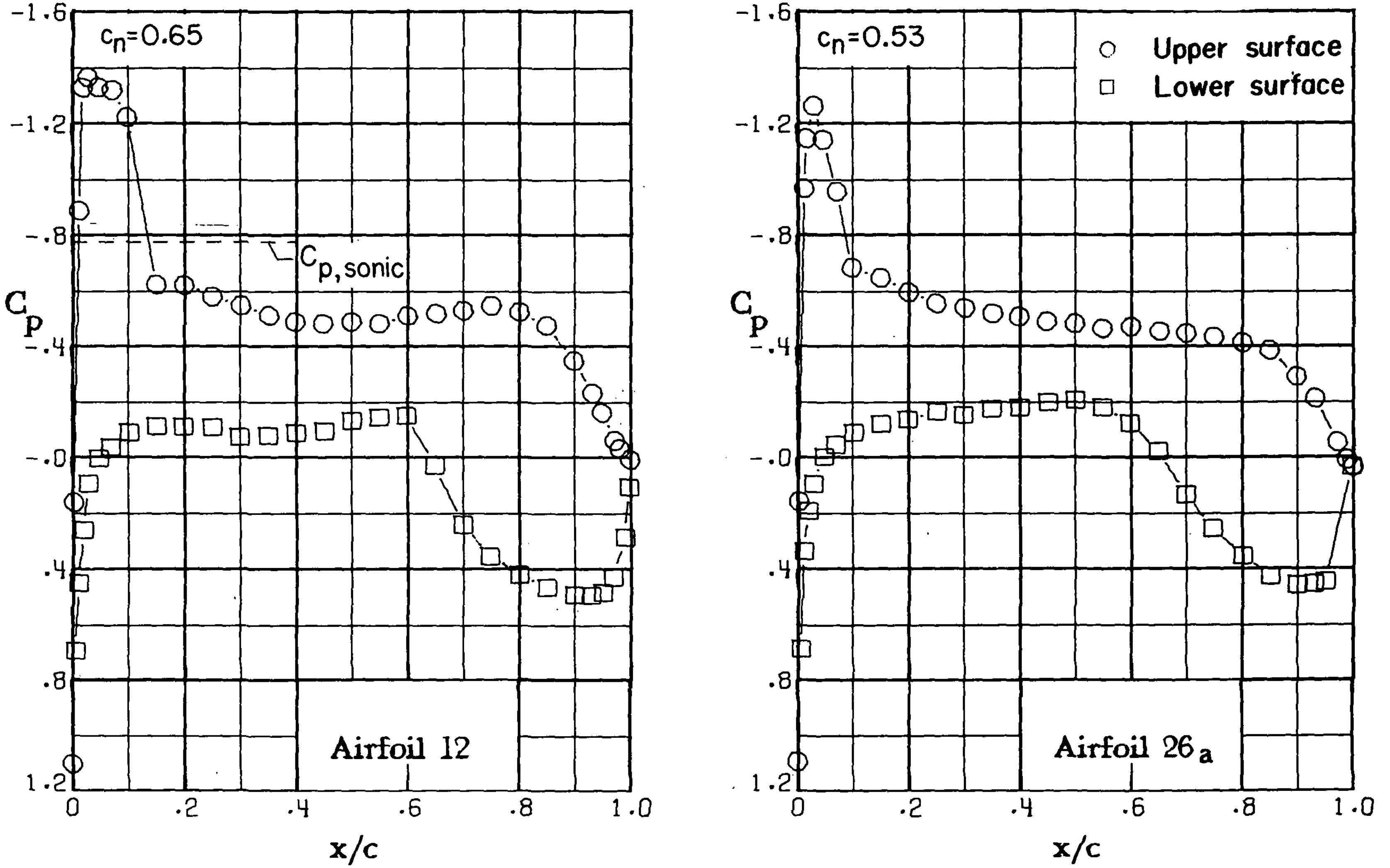
(a) M = 0.70; α = -0.5°.

Figure 14.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.70.



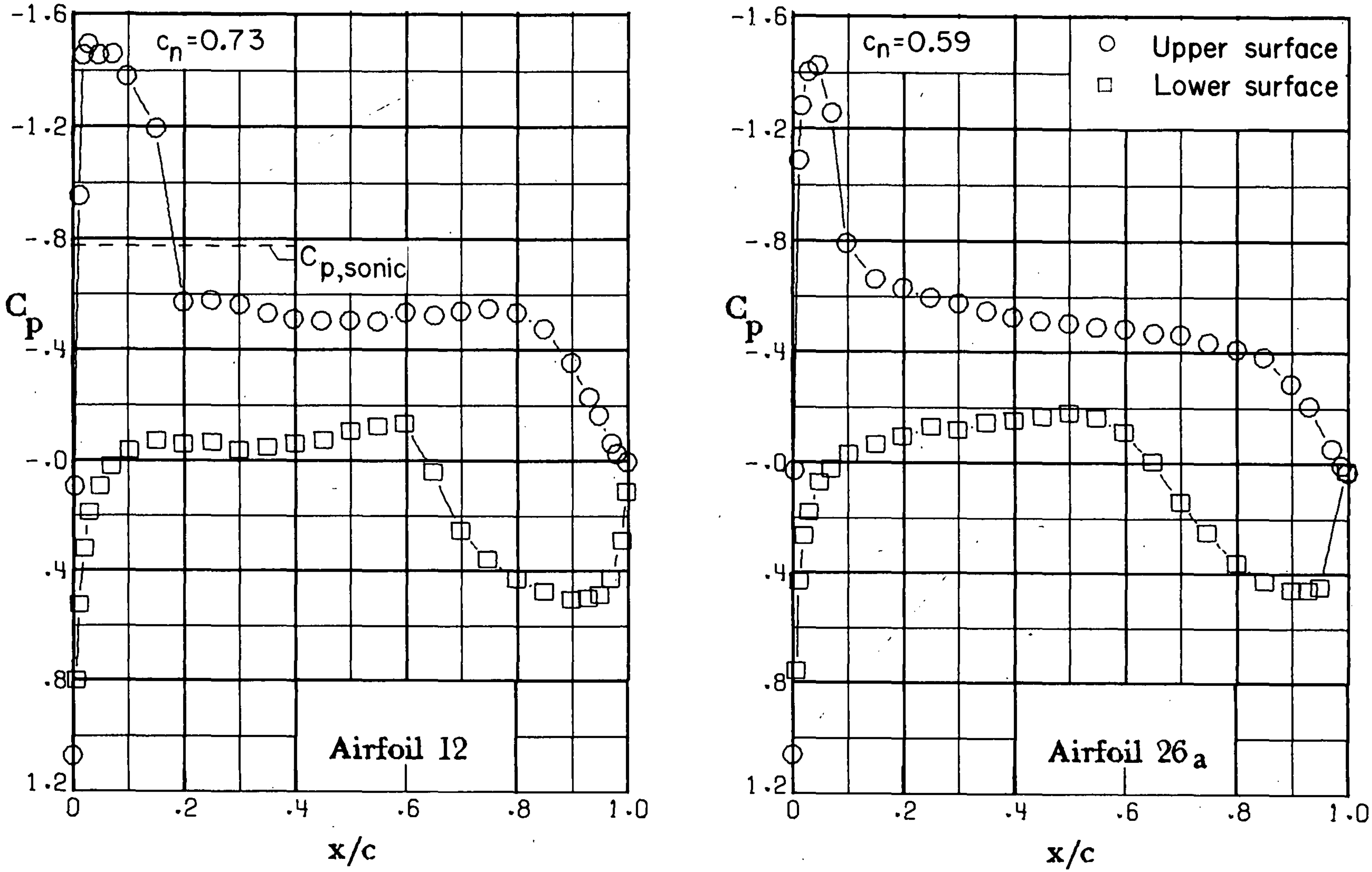
(b) $M = 0.70$; $\alpha = 1.0^{o}$.

Figure 14.- Continued.



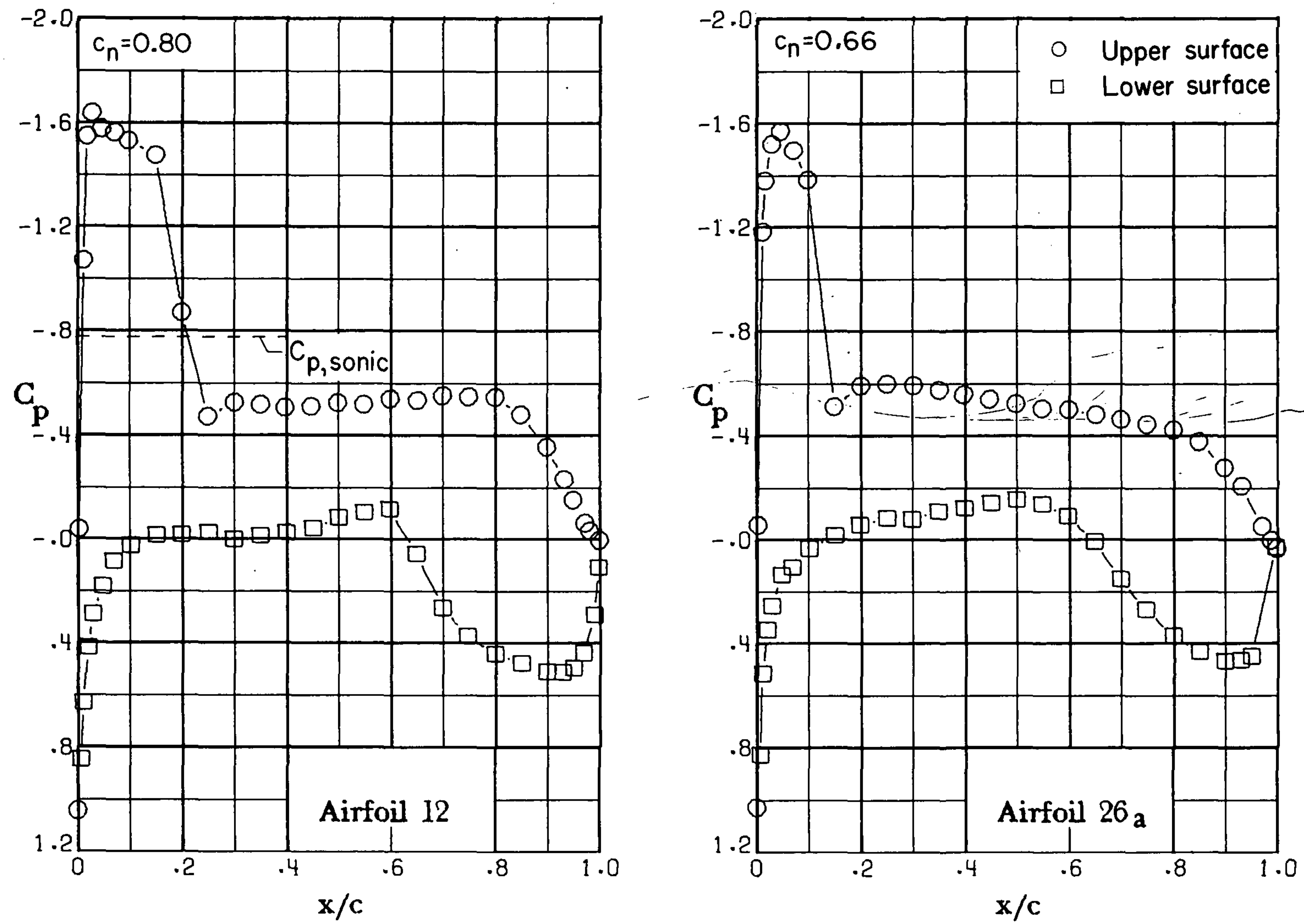
(c) M = 0.70; α = 1.5°.

Figure 14.- Continued.



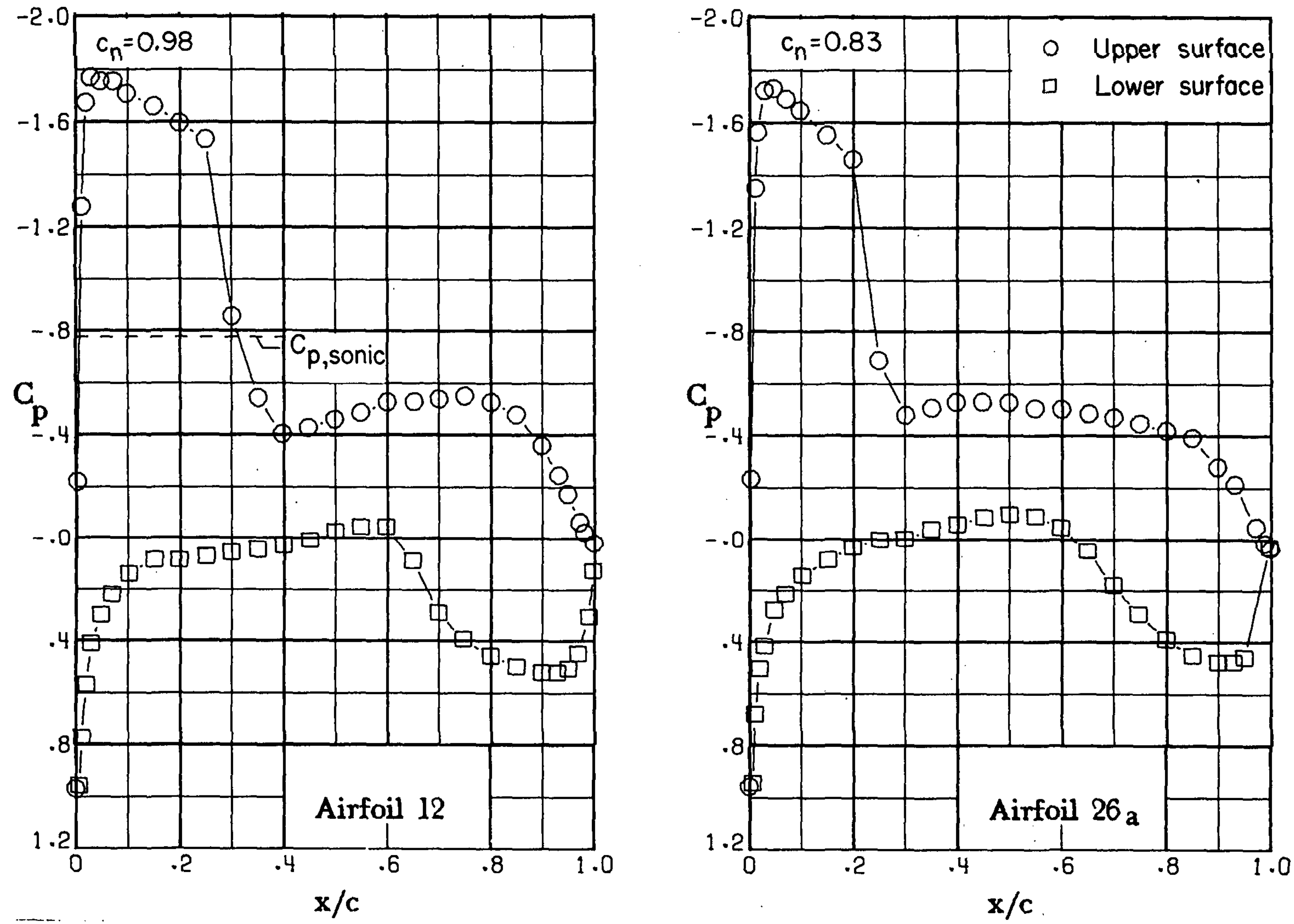
(d) $M = 0.70$; $\alpha = 2.0^{\circ}$.

Figure 14.- Continued.



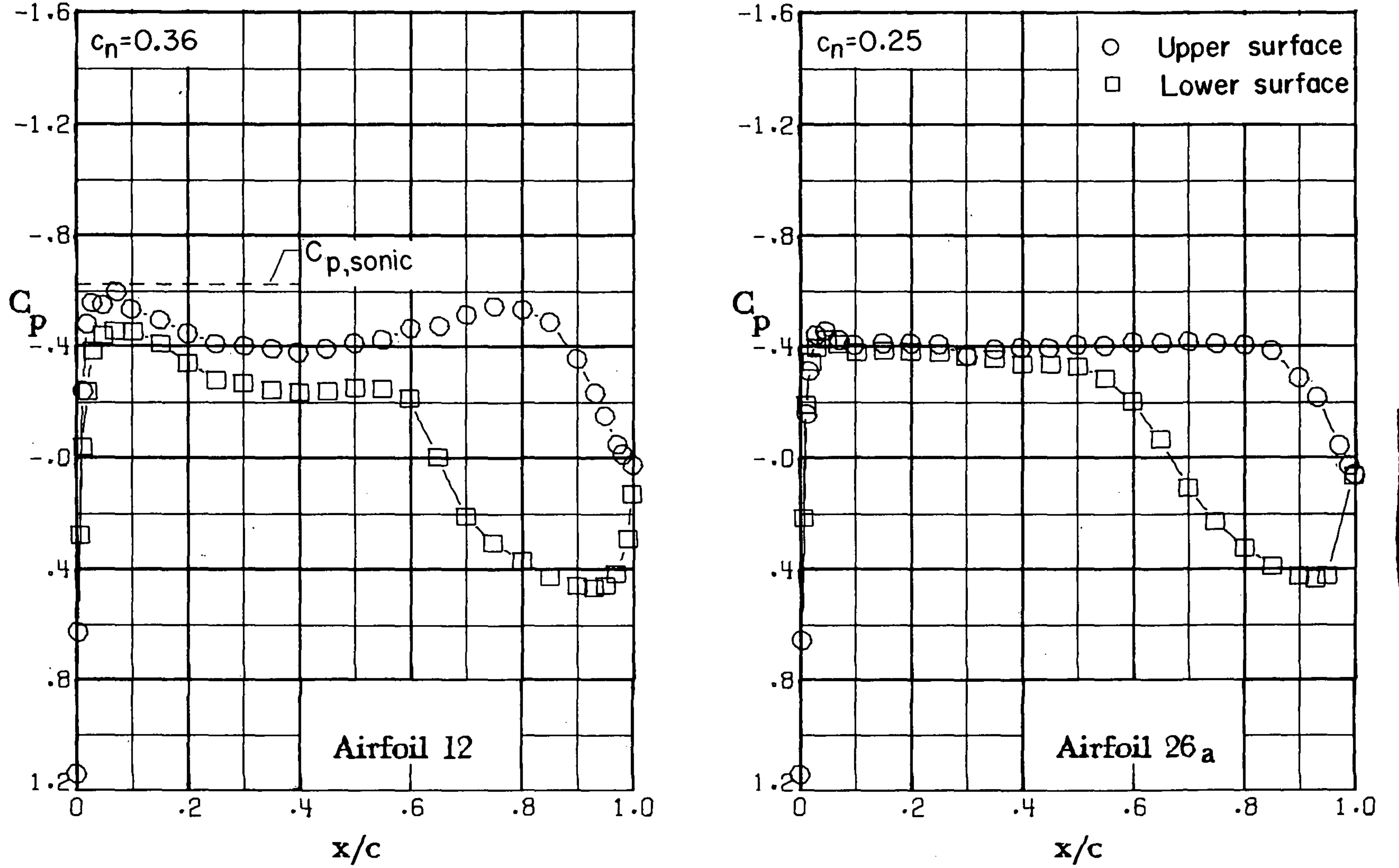
(e) M = 0.70; $\alpha = 2.5^o$.

Figure 14.- Continued.



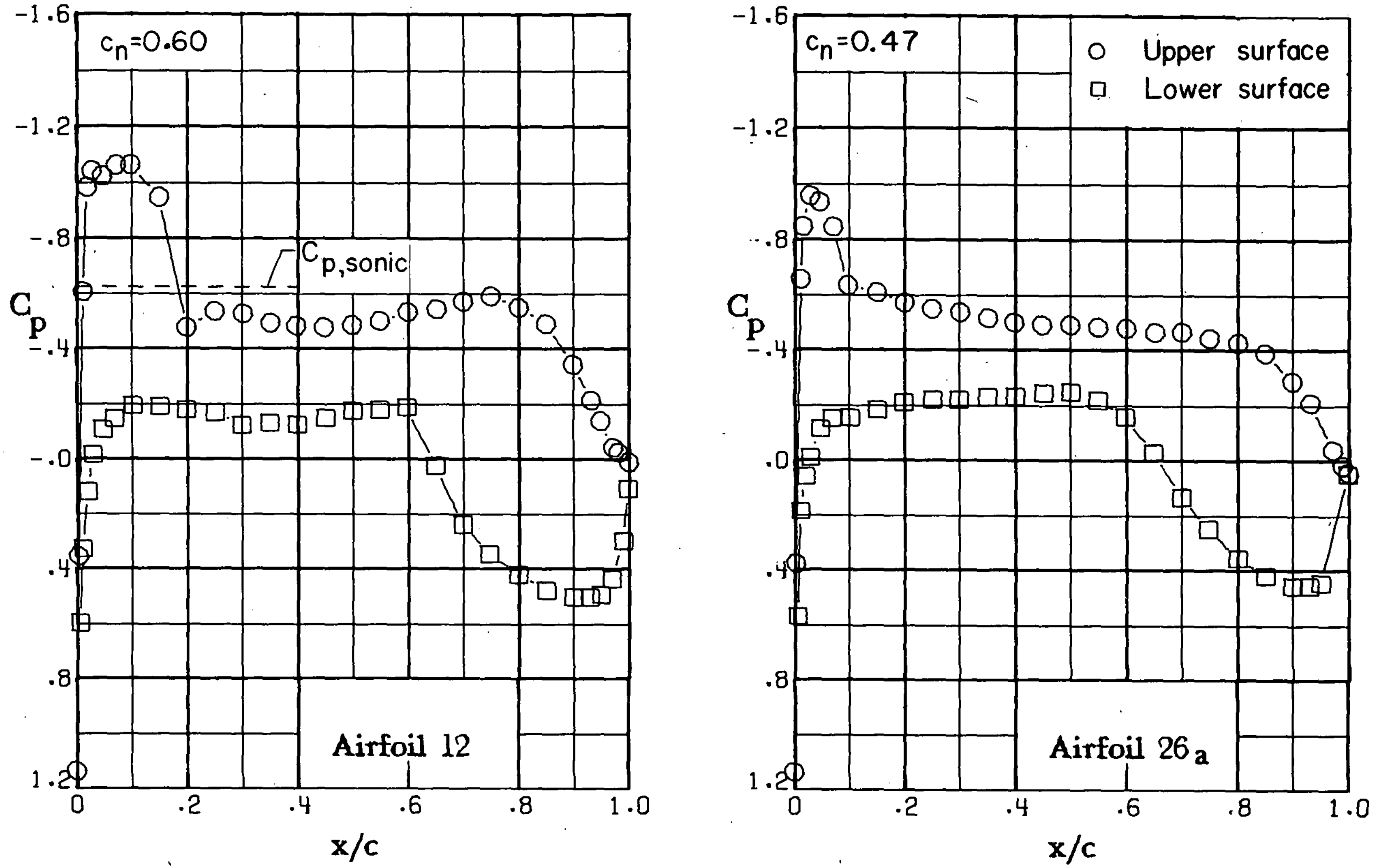
(f) M = 0.70; α = 3.5°.

Figure 14.- Concluded.



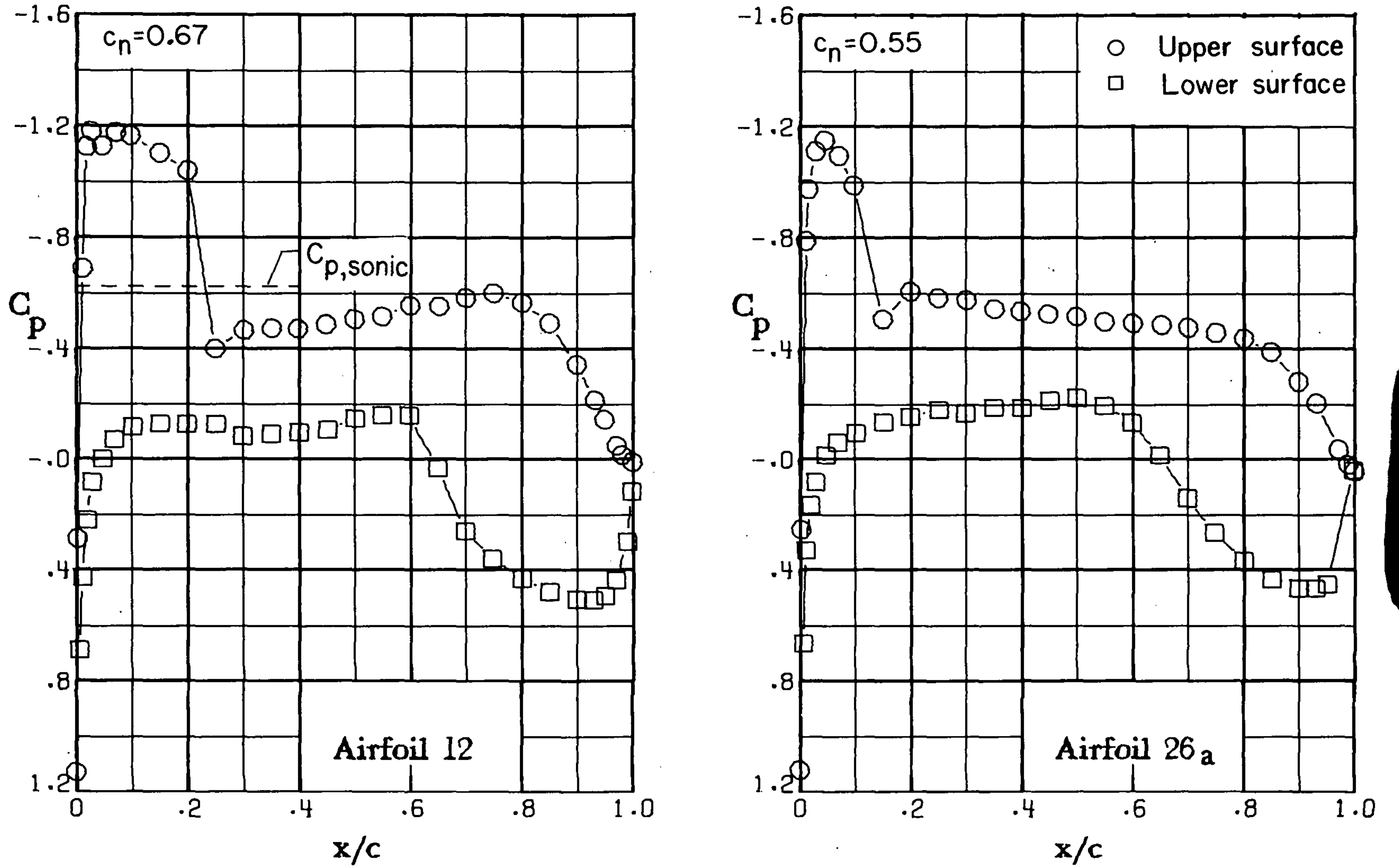
(a) M = 0.74; α = -0.5°.

Figure 15.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.74.



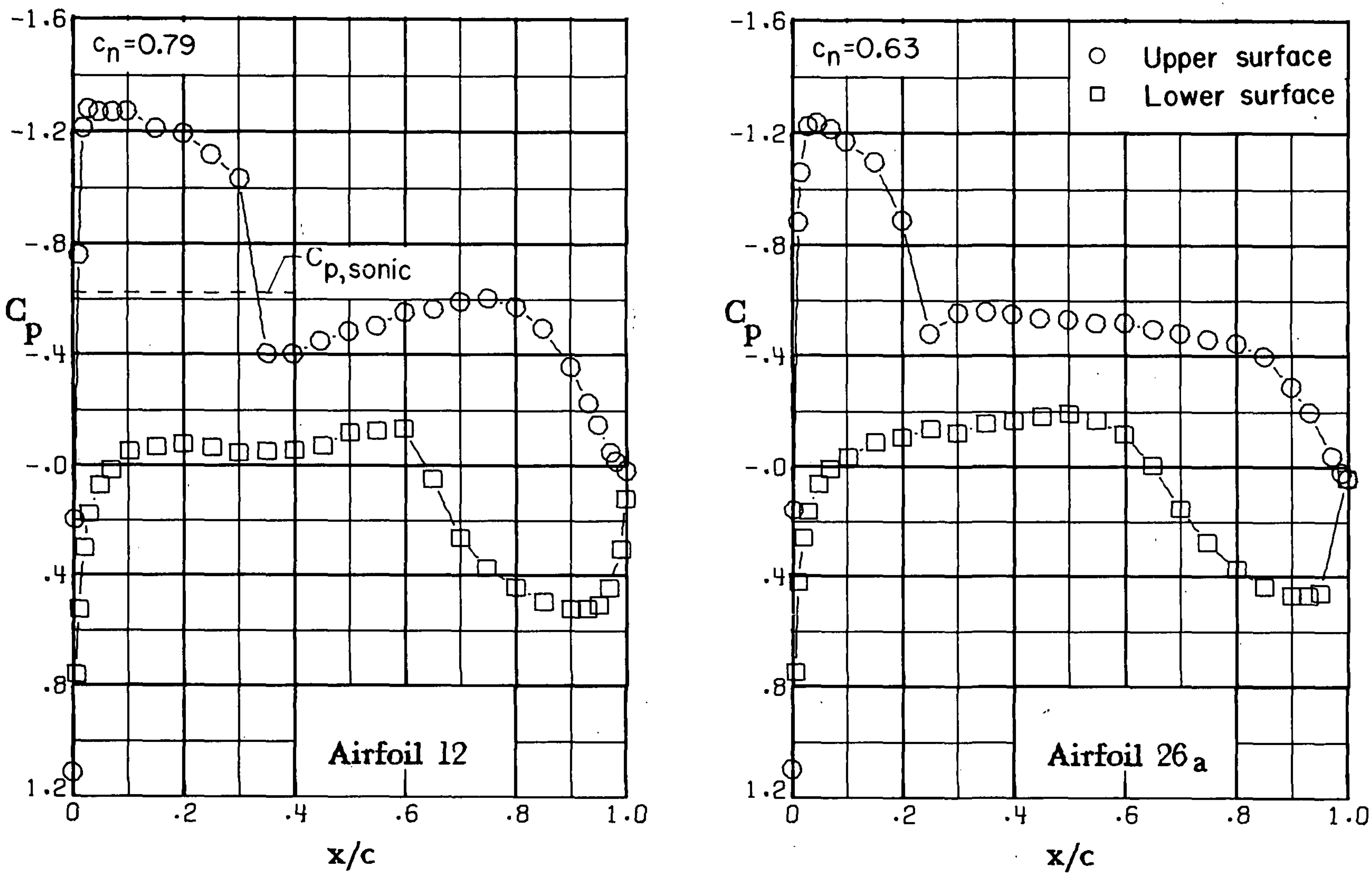
(b) $M = 0.74$; $\alpha = 1.0^{o}$.

Figure 15.- Continued.



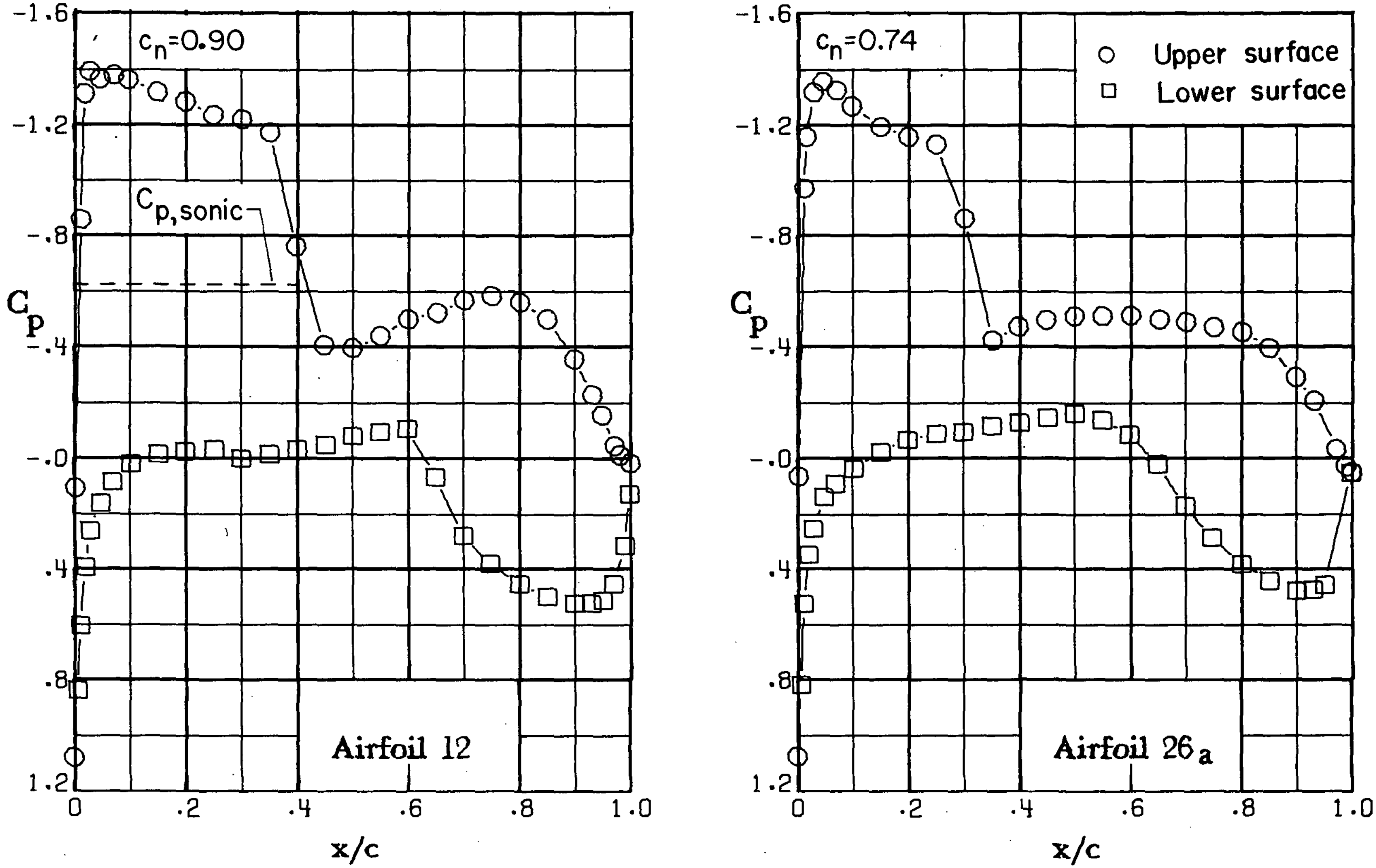
(c) M = 0.74; α = 1.5°.

Figure 15.- Continued.



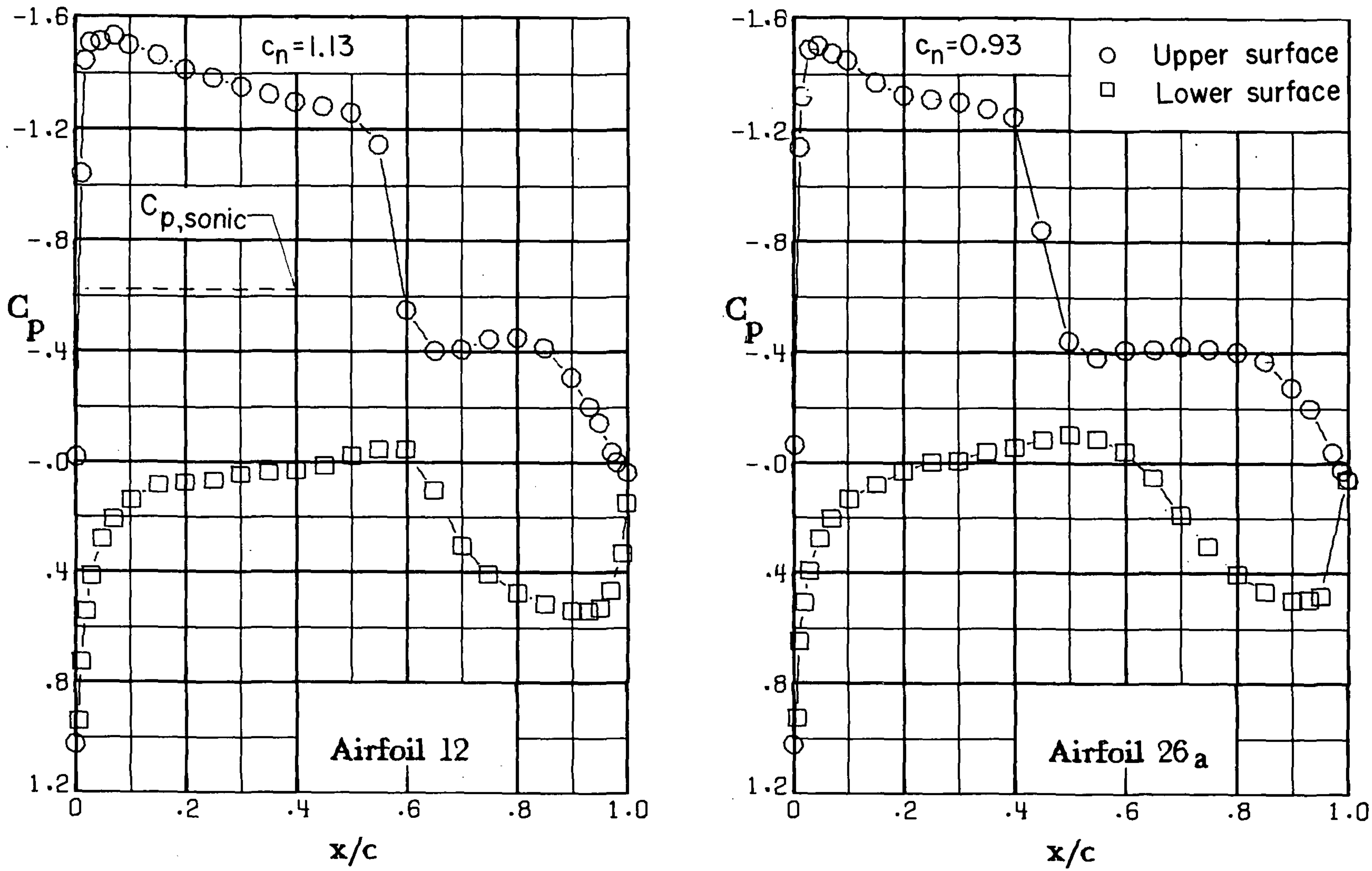
(d) M = 0.74; α = 2.0°.

Figure 15.- Continued.



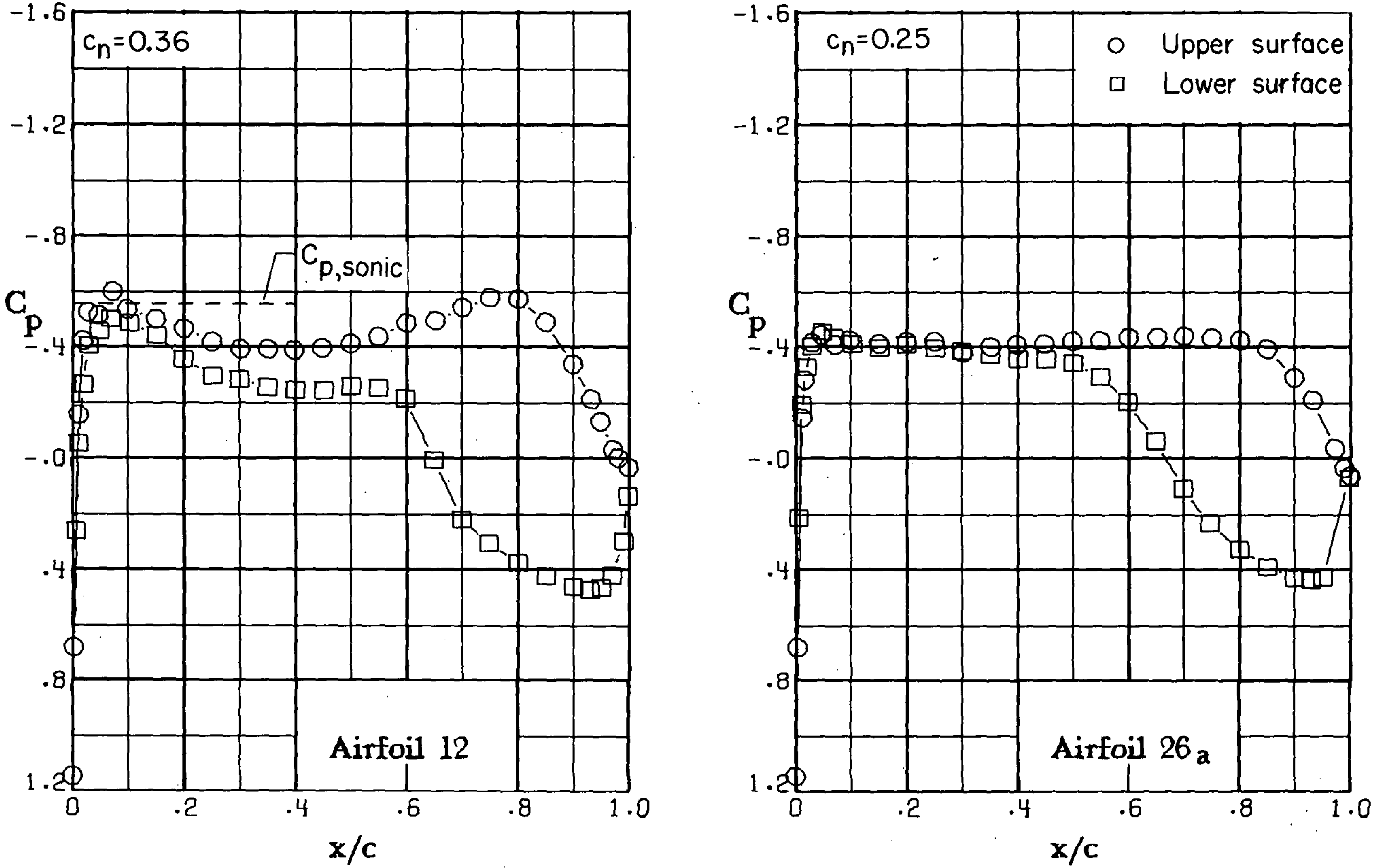
(e) M = 0.74; $\alpha = 2.5^{\circ}$.

Figure 15.- Continued.



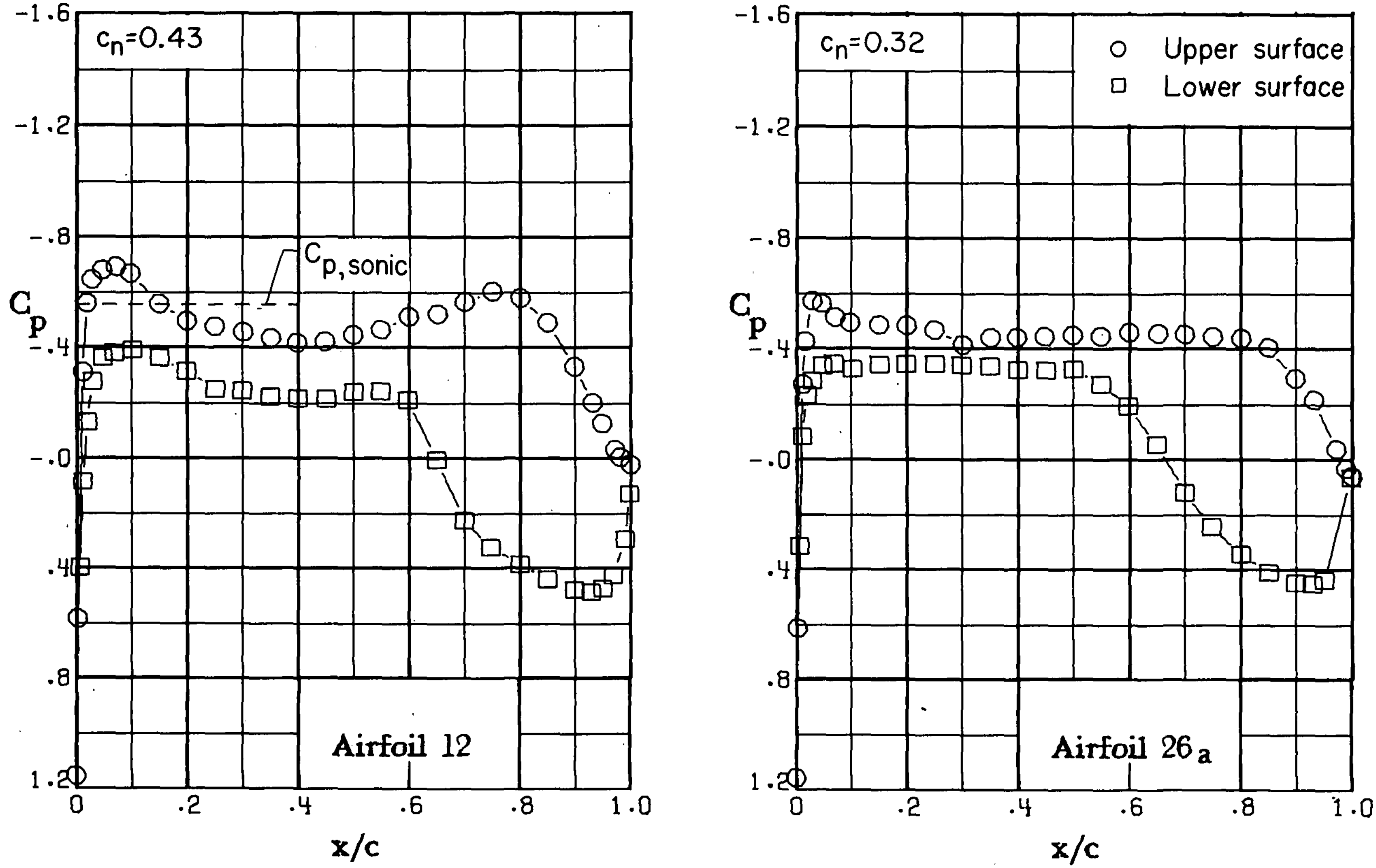
(f) M = 0.74; $\alpha = 3.5^{\circ}$.

Figure 15.- Concluded.



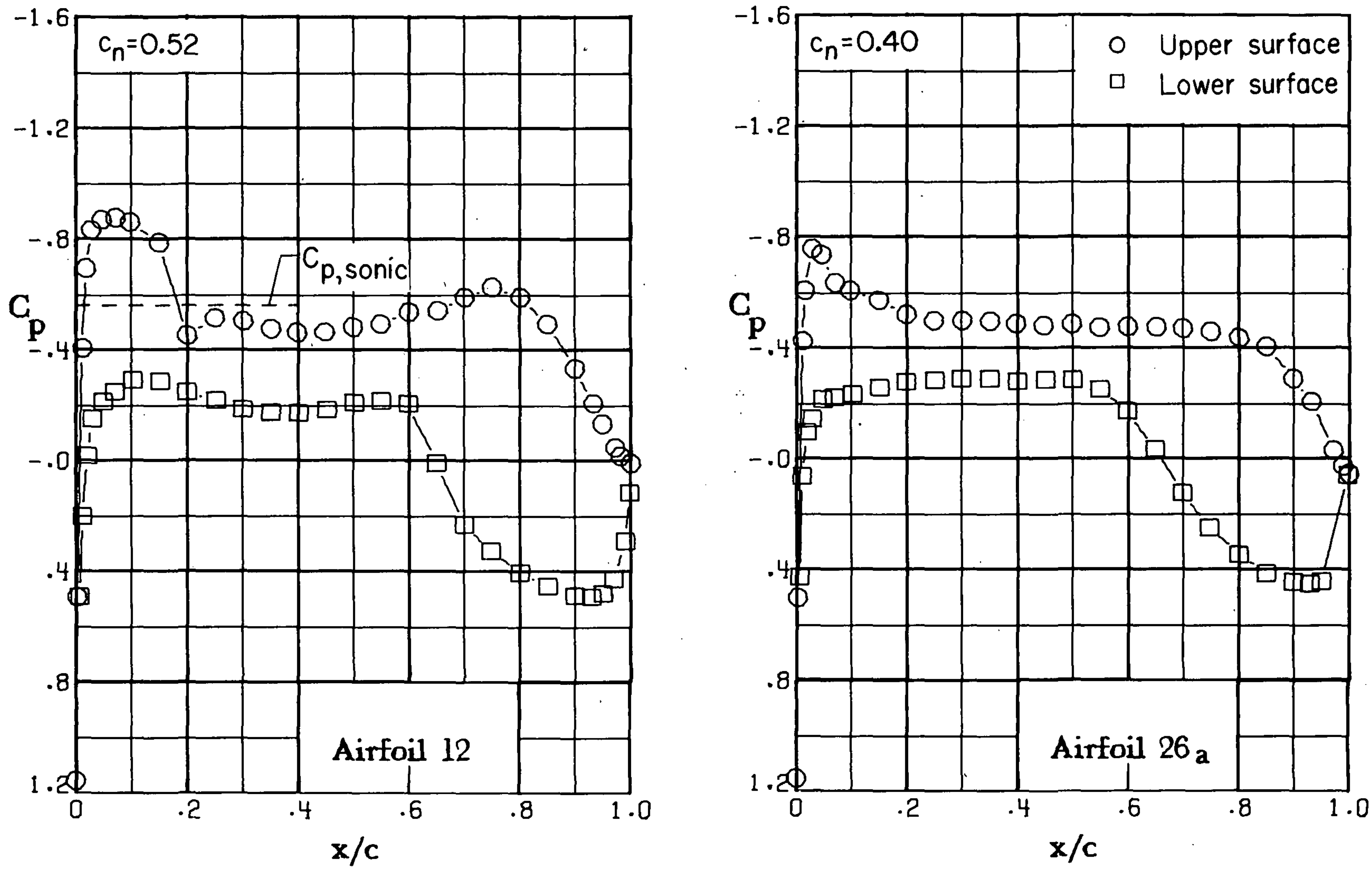
(a) M = 0.76; α = -0.5°.

Figure 16.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.76.



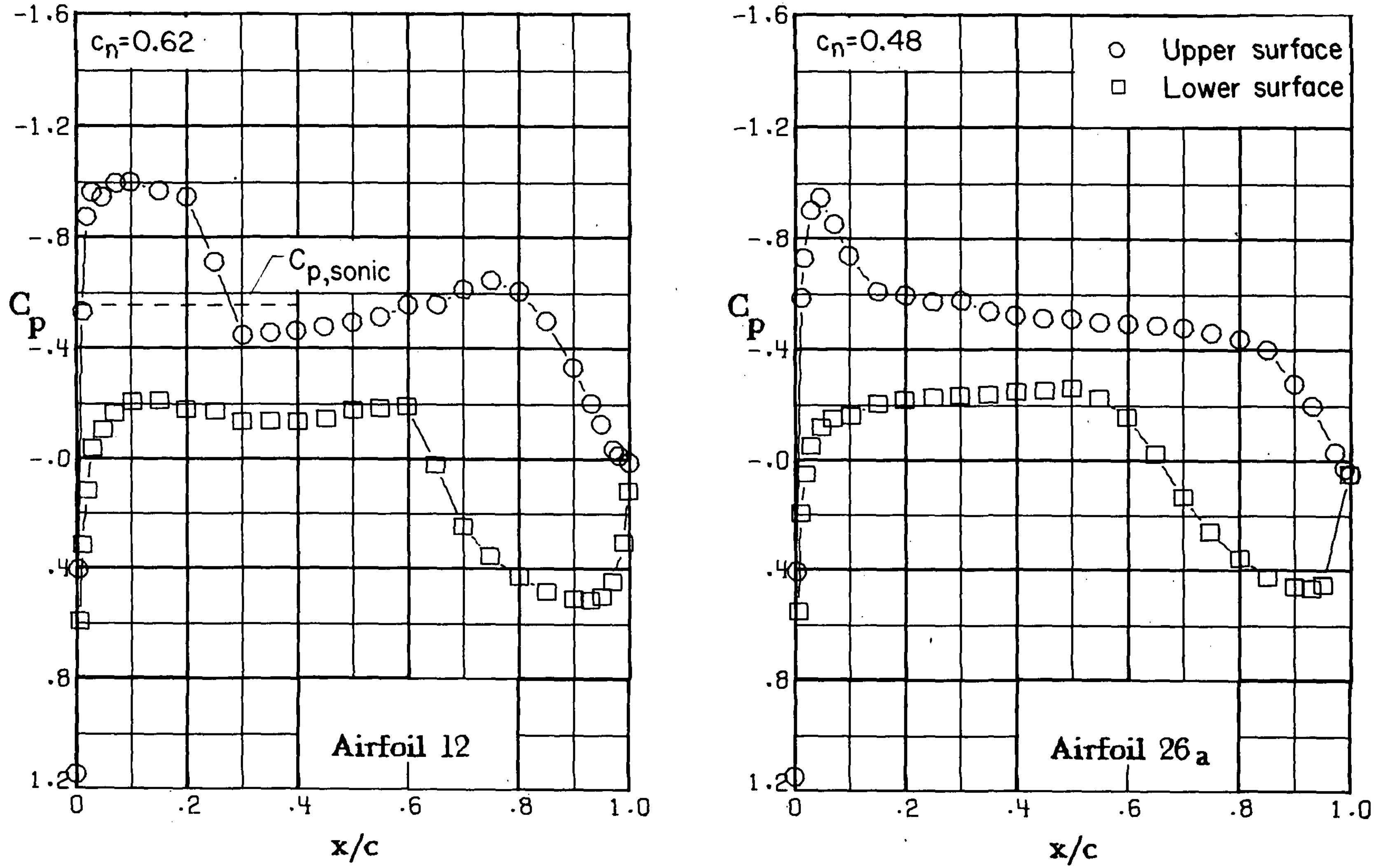
(b) M = 0.76; $\alpha = 0.0^{o}$.

Figure 16.- Continued.



(c) M = 0.76; $\alpha = 0.5^o$.

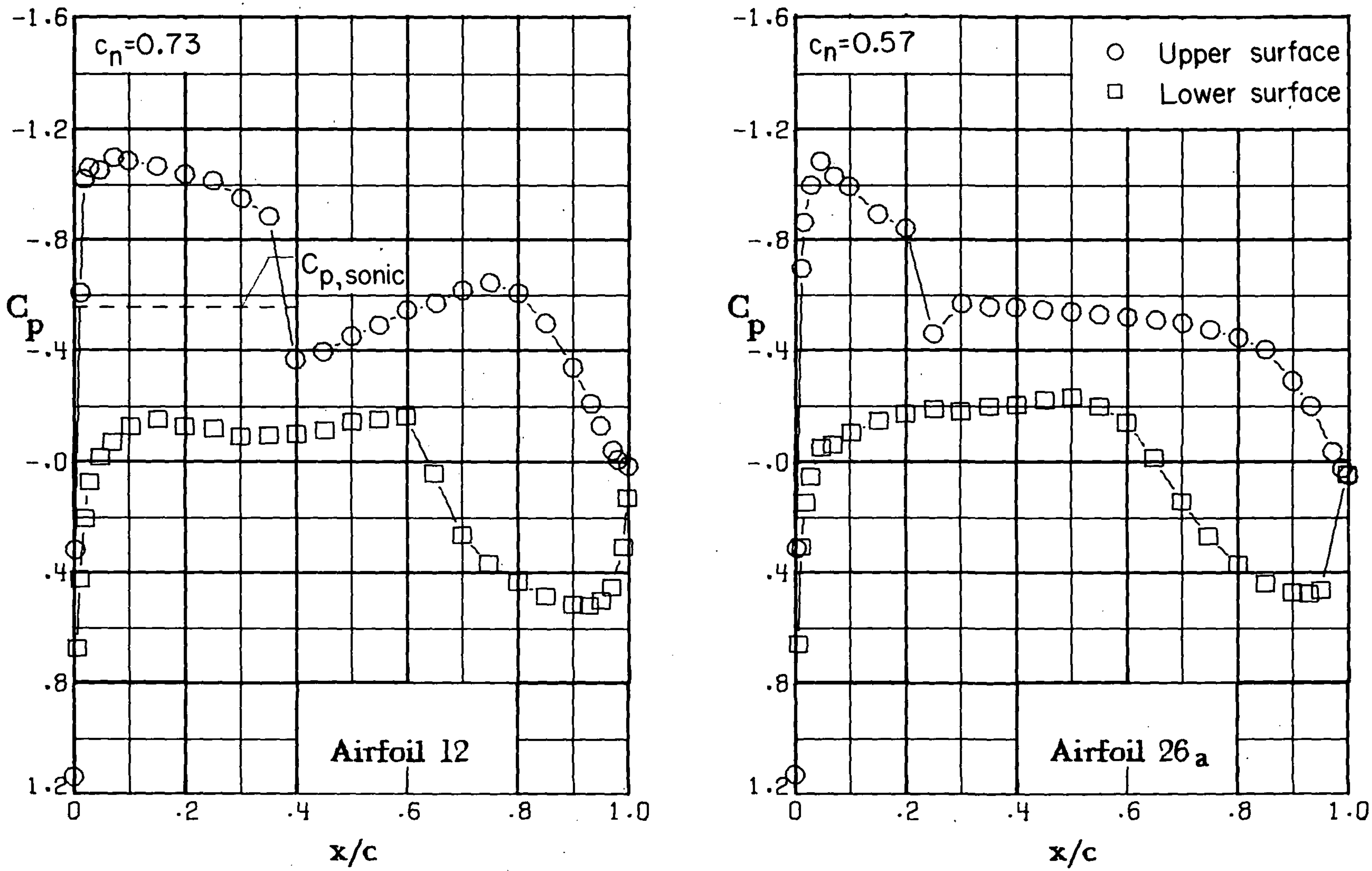
Figure 16.- Continued.



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

Figure 16.- Continued.

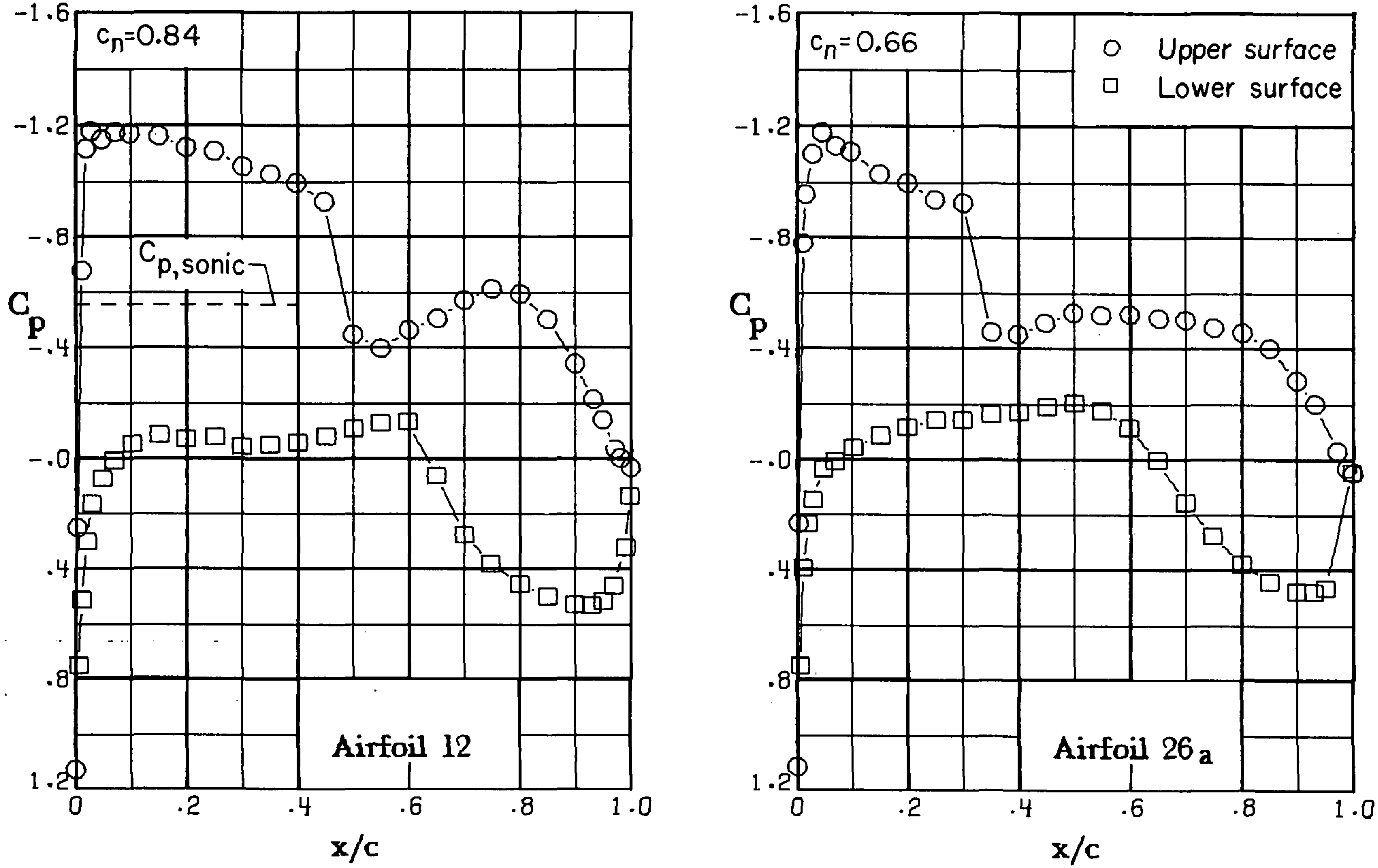
[redacted]



(e) M = 0.76; α = 1.5°.

Figure 16.- Continued.

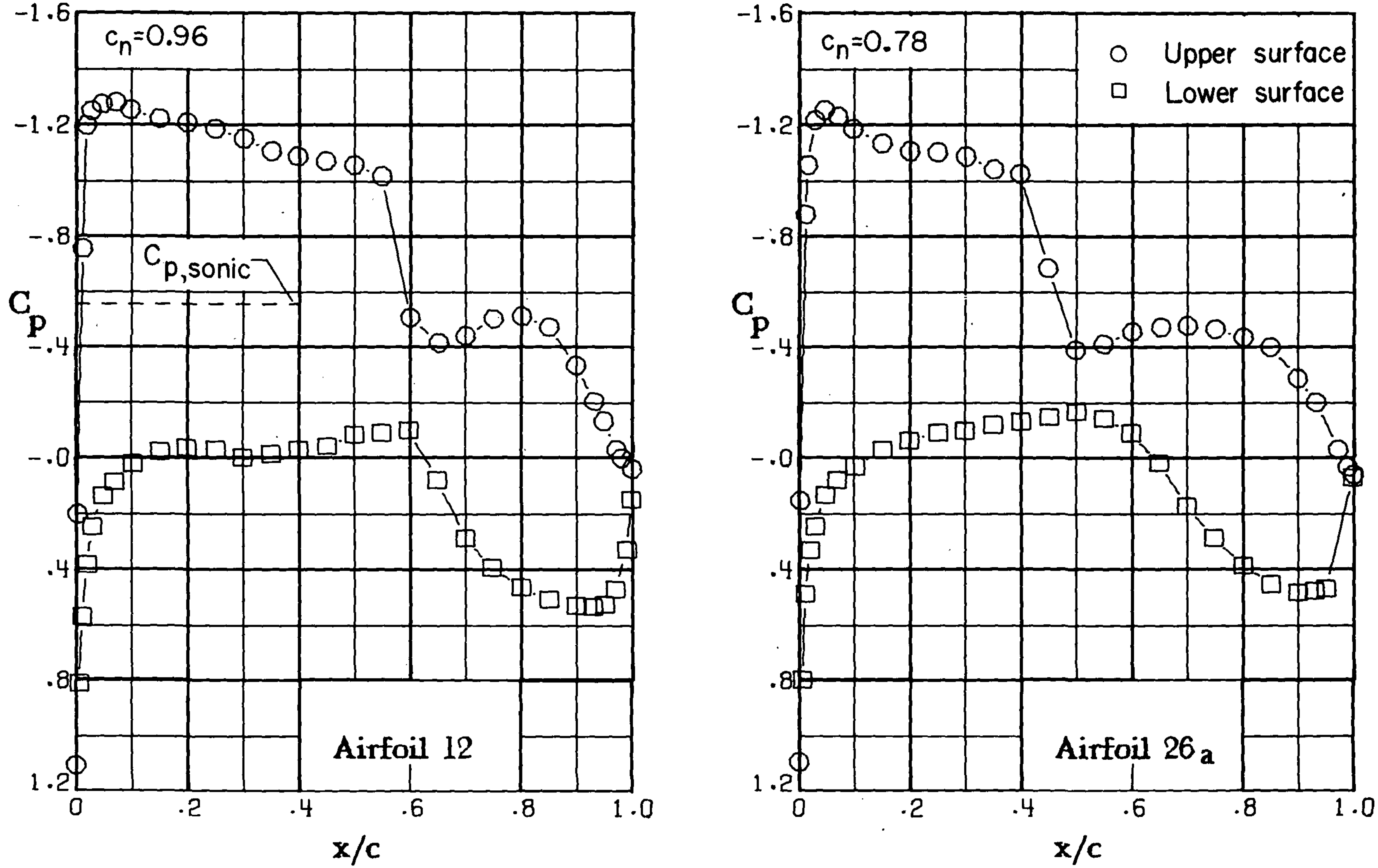
[redacted]



(f) M = 0.76; α = 2.0°.

Figure 16.- Continued.

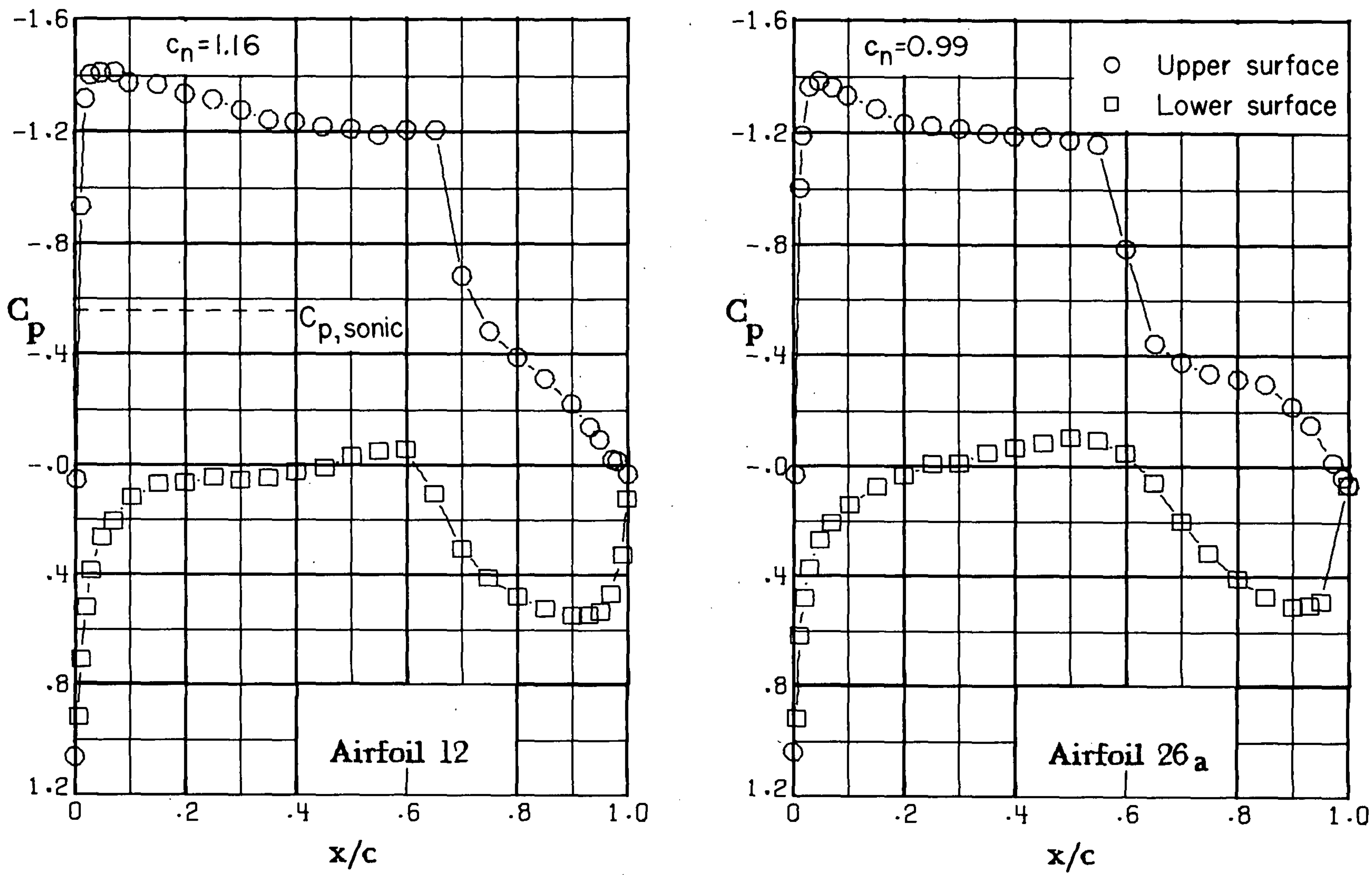
[redacted]



(g) M = 0.76; $\alpha = 2.5^{o}$.

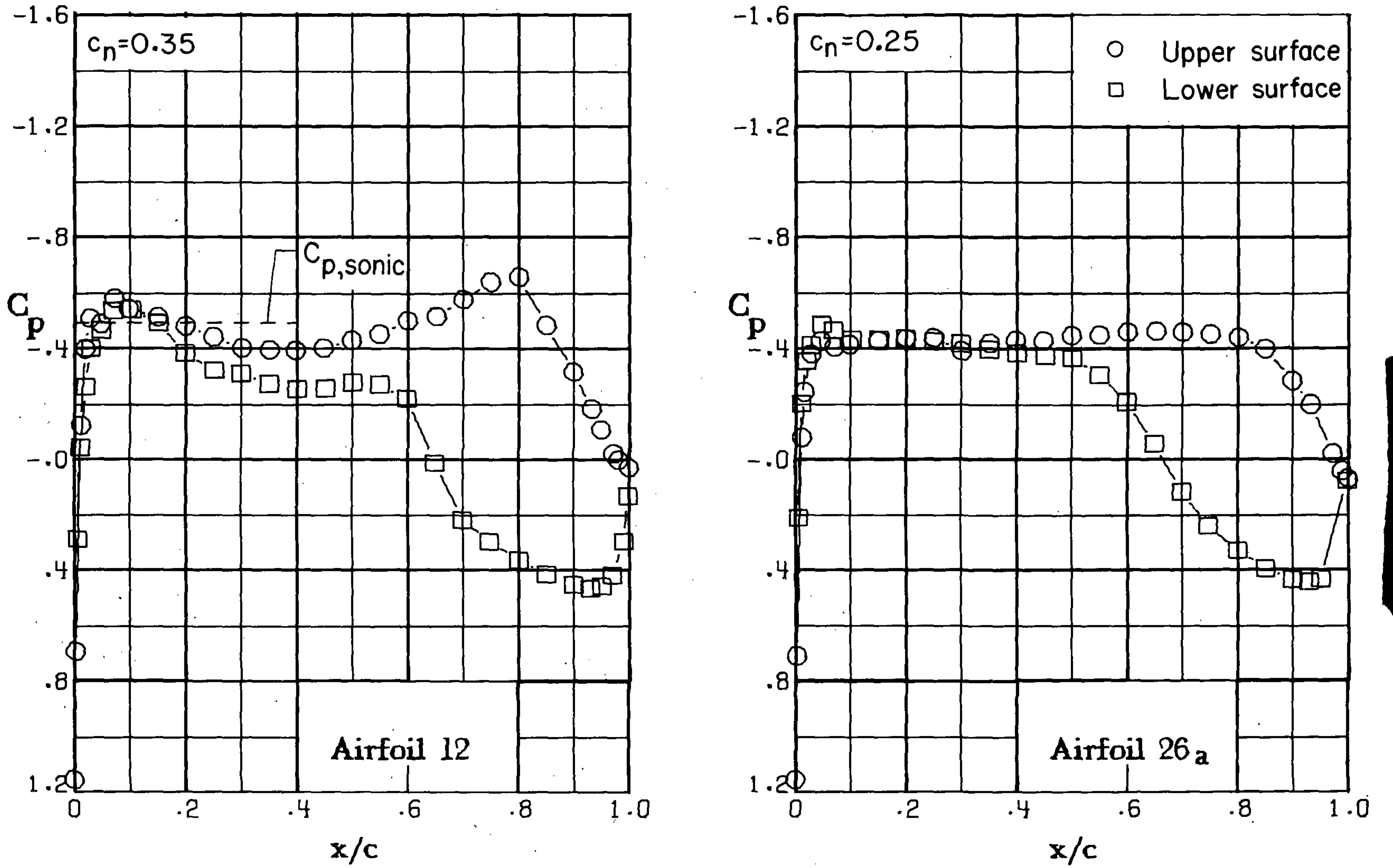
Figure 16.- Continued.

[redacted]



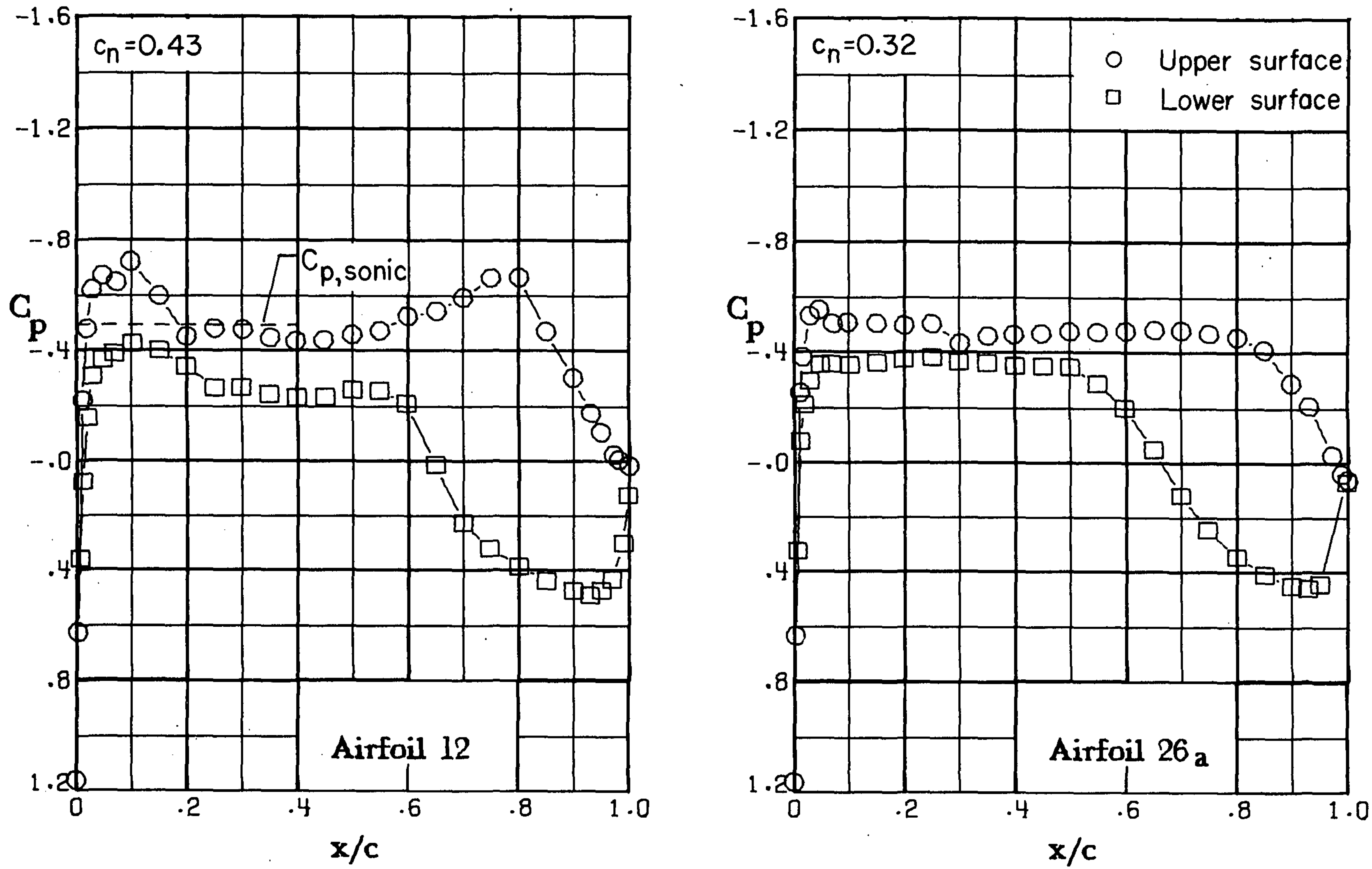
(h) M = 0.76; α = 3.5°.

Figure 16.- Concluded.



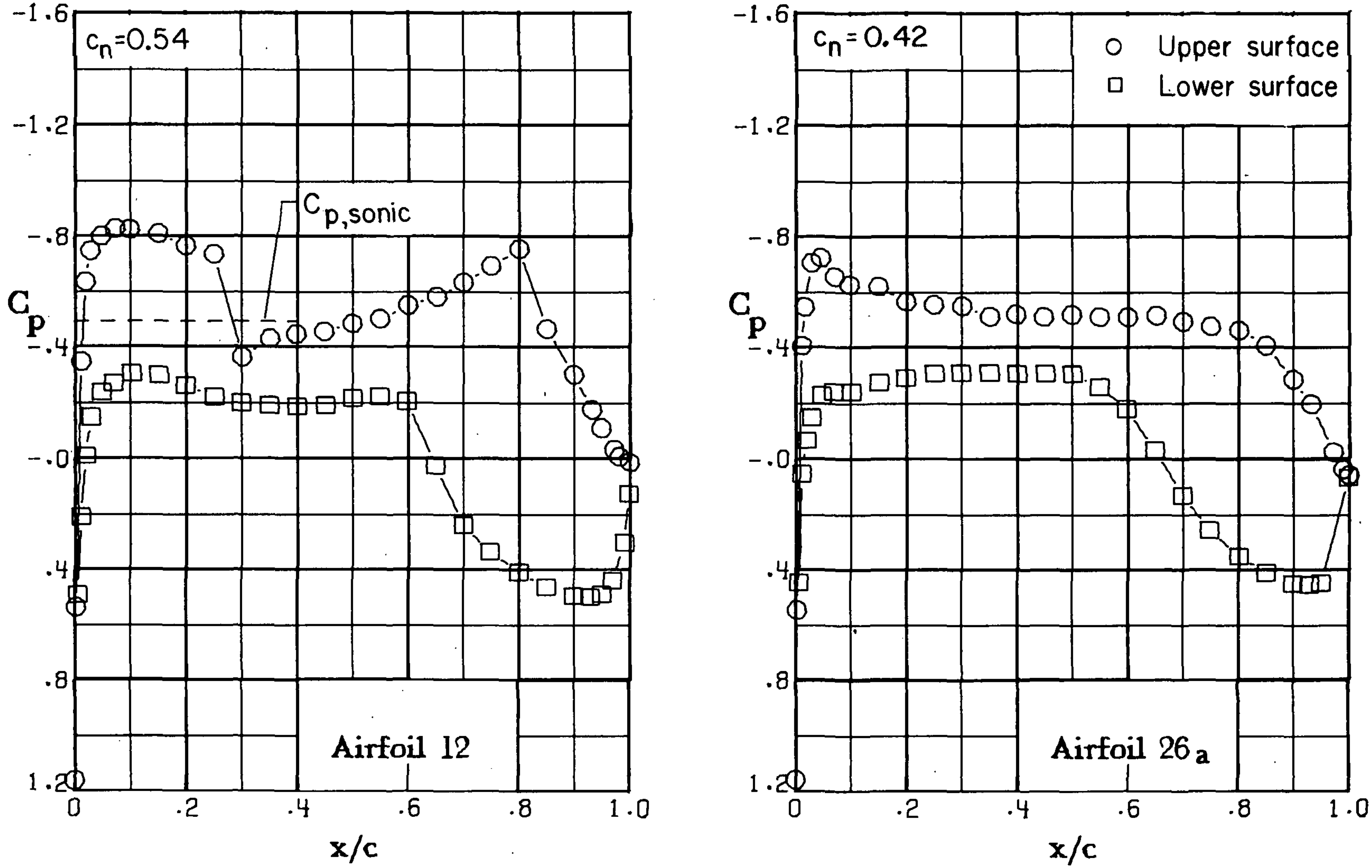
(a) M = 0.78; α = -0.5°.

Figure 17.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.78.



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

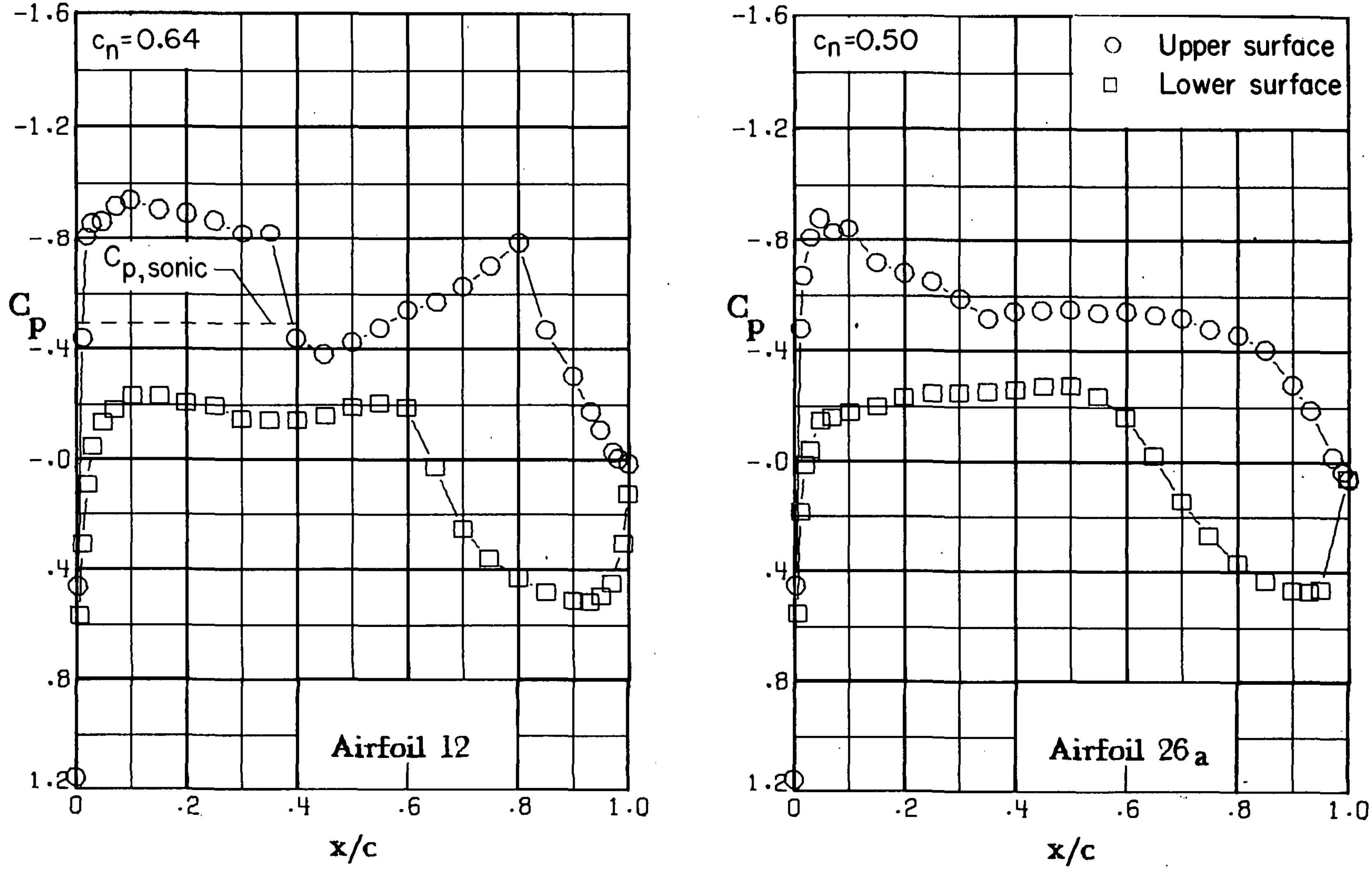
Figure 17.- Continued.



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

Figure 17.- Continued.

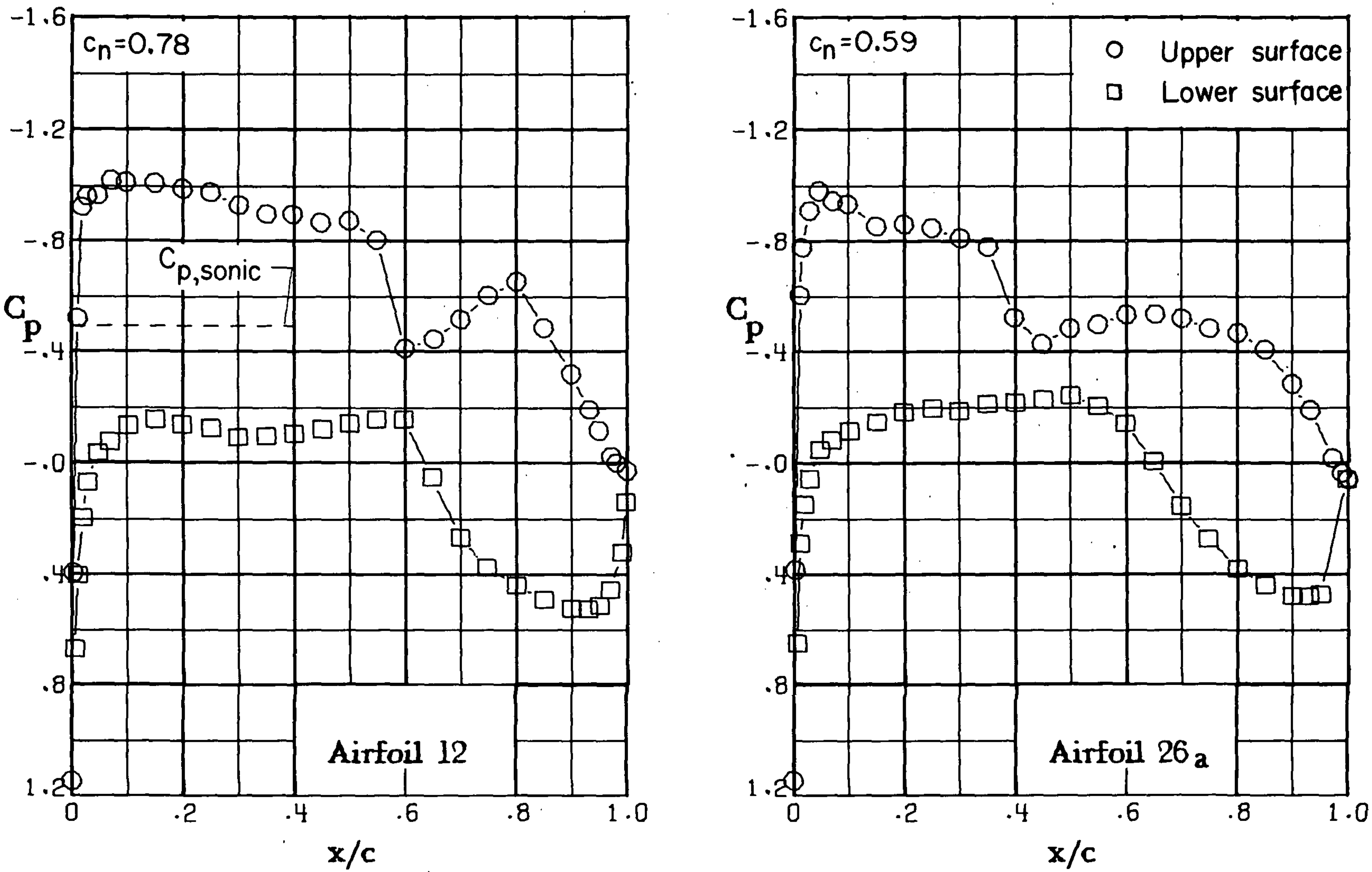
[redacted]



(d) M = 0.78; $\alpha = 1.0^{\circ}$.

Figure 17.- Continued.

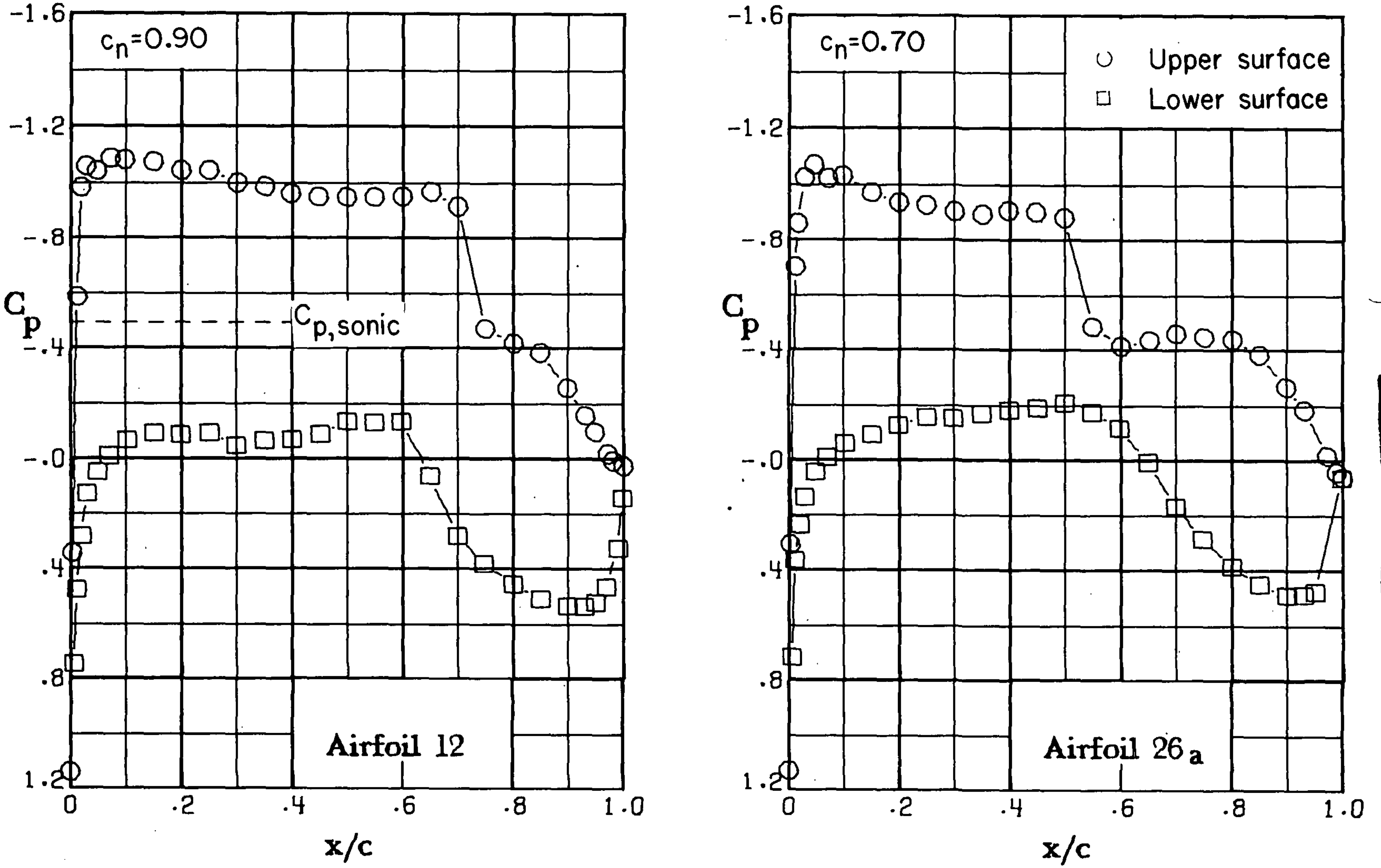
[redacted]



(e) M = 0.78; α = 1.5°.

Figure 17.- Continued.

[redacted]

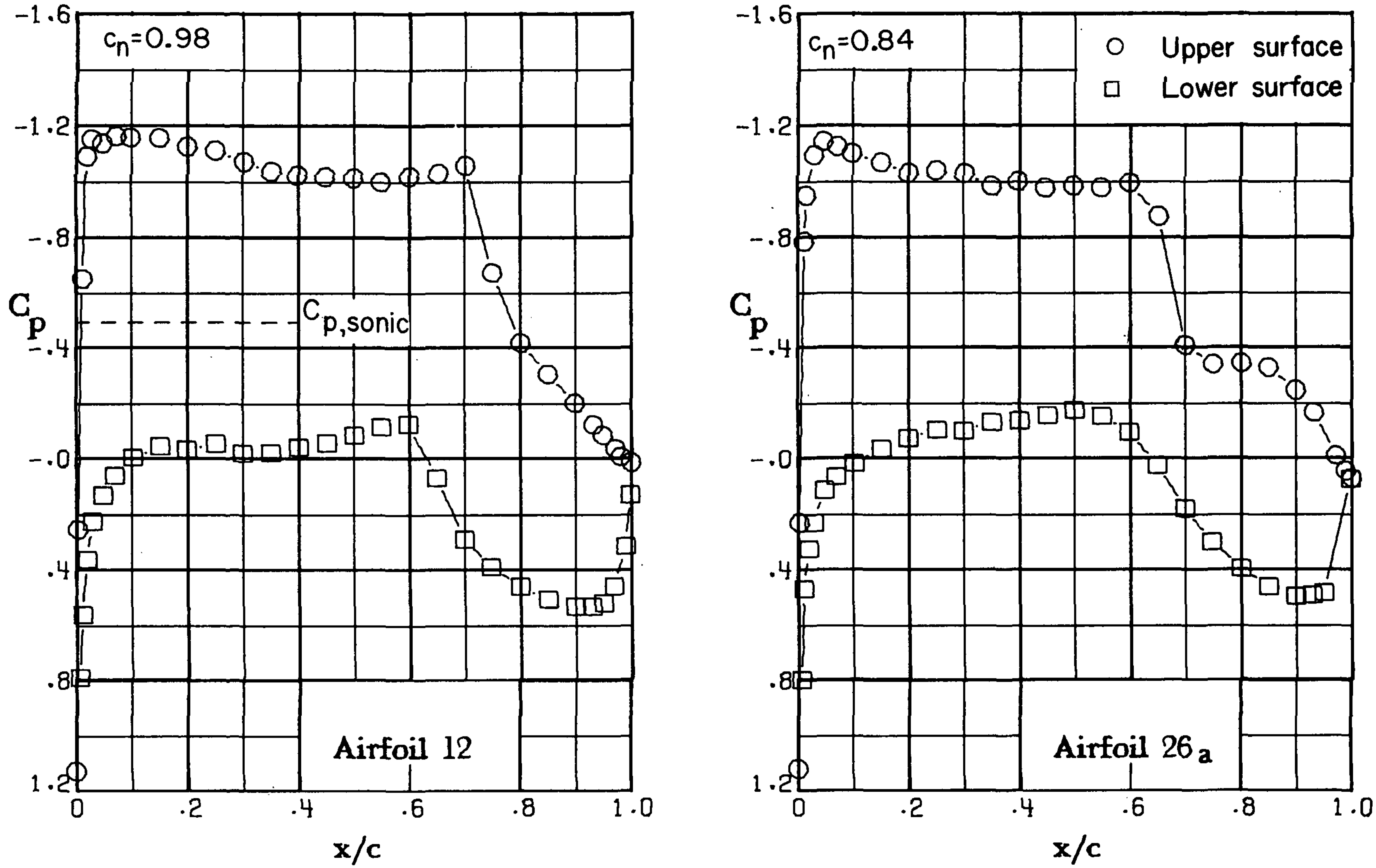


(f) $M = 0.78$; $\alpha = 2.0^{o}$.

Figure 17.- Continued.

[redacted]

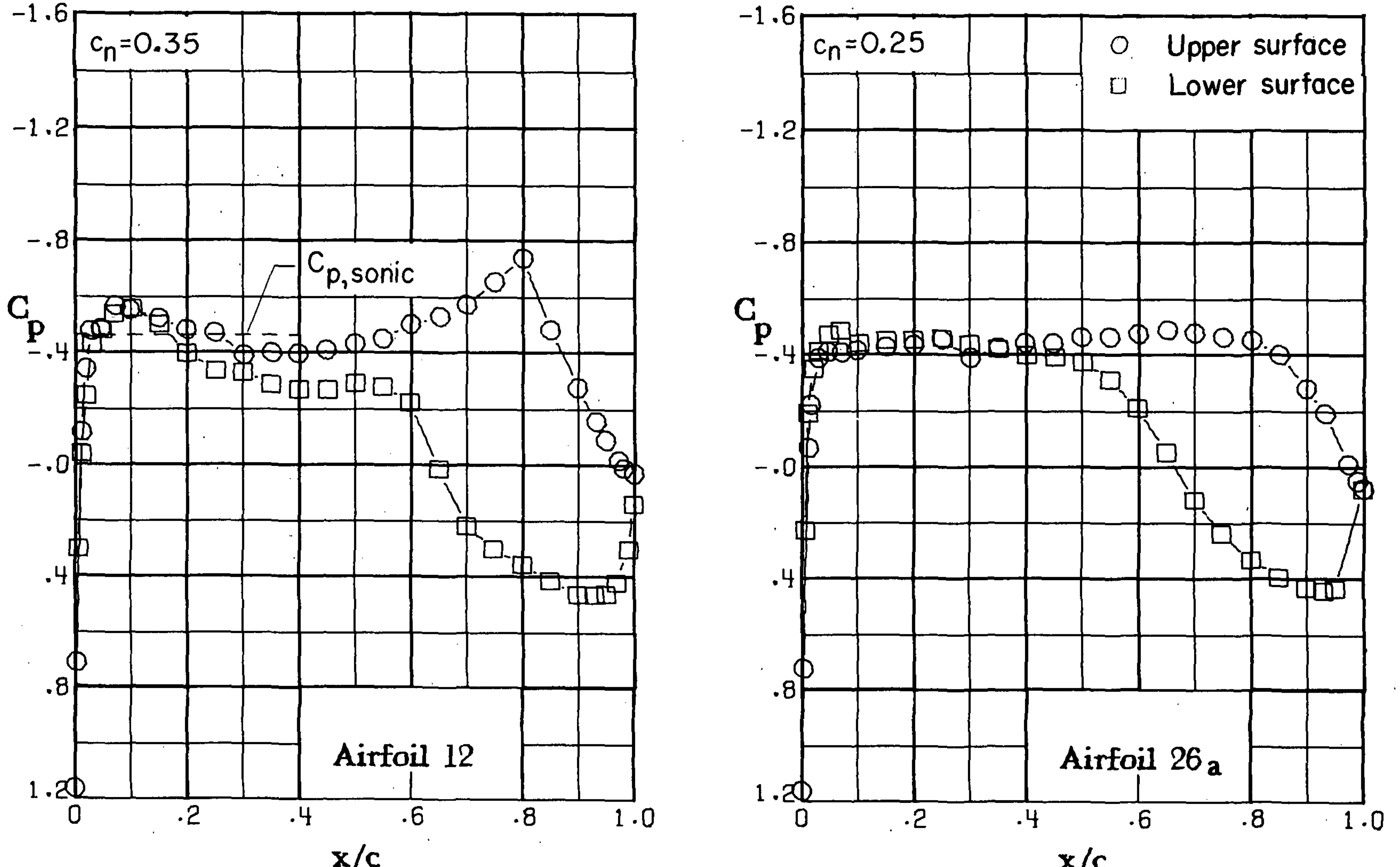
[redacted]



(g) M = 0.78; α = 2.5°.

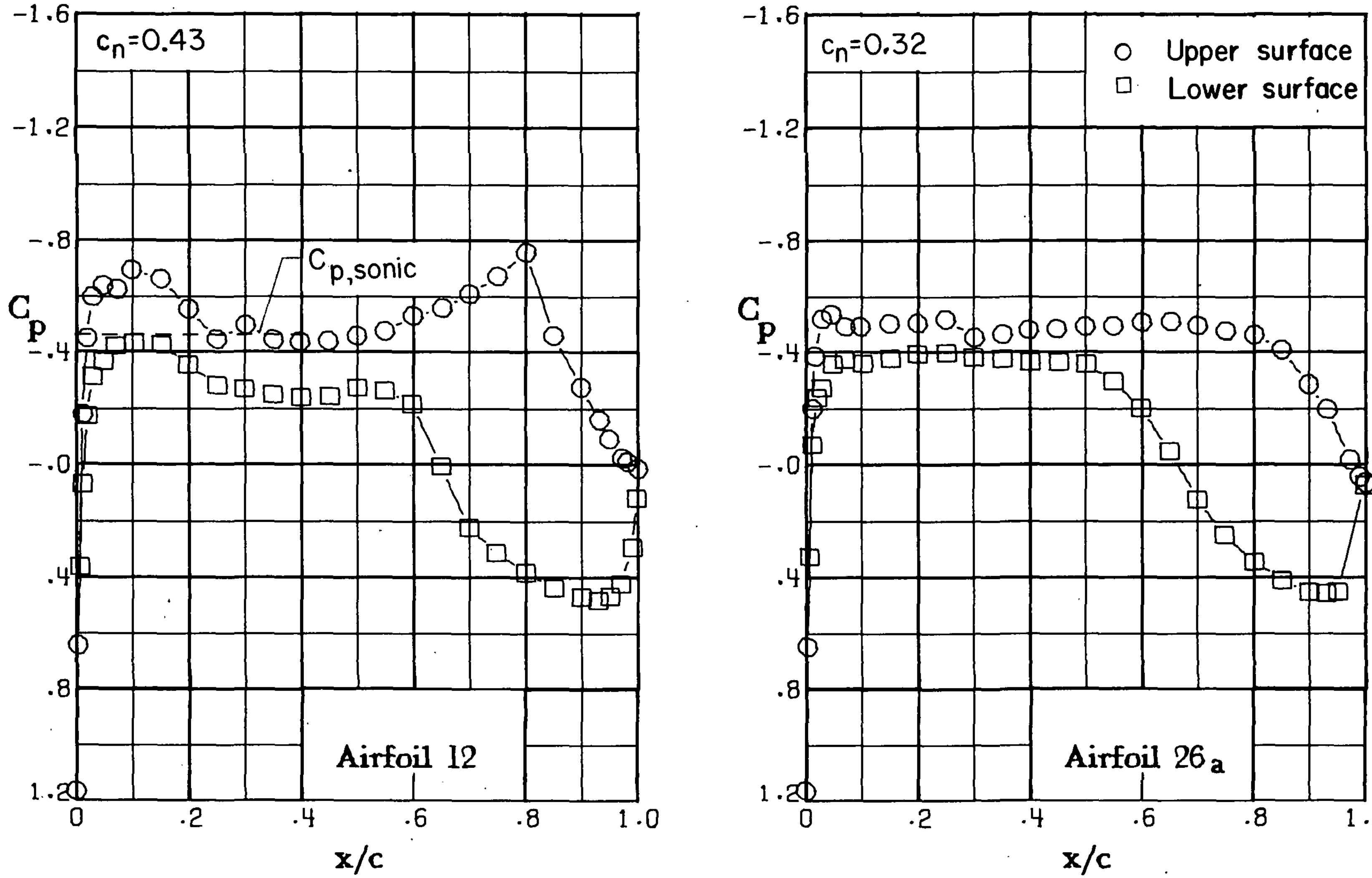
Figure 17.- Concluded.

[redacted]



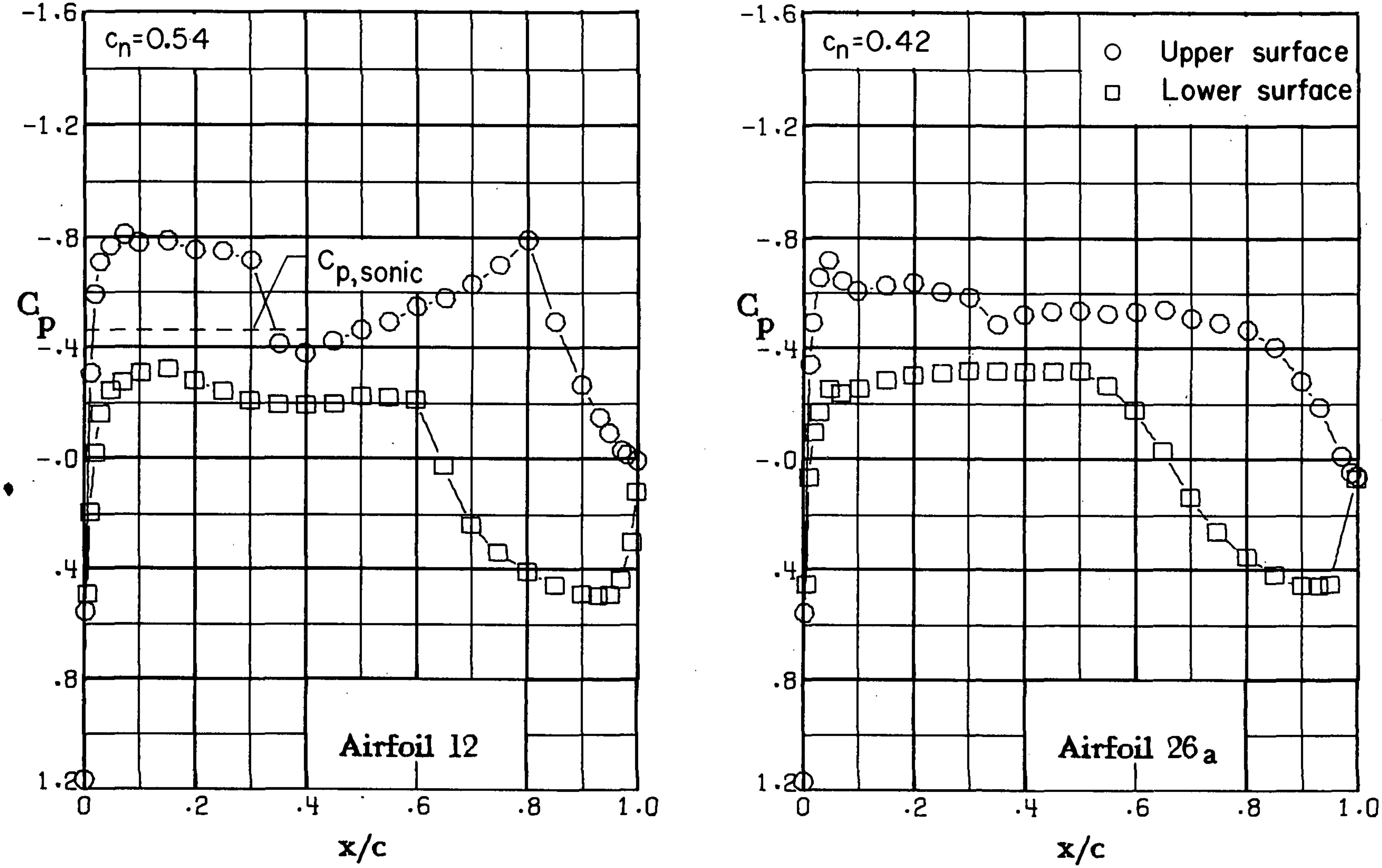
(a) M = 0.79; α = -0.5°.

Figure 18.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.79.



(b) M = 0.79; α = 0.0°.

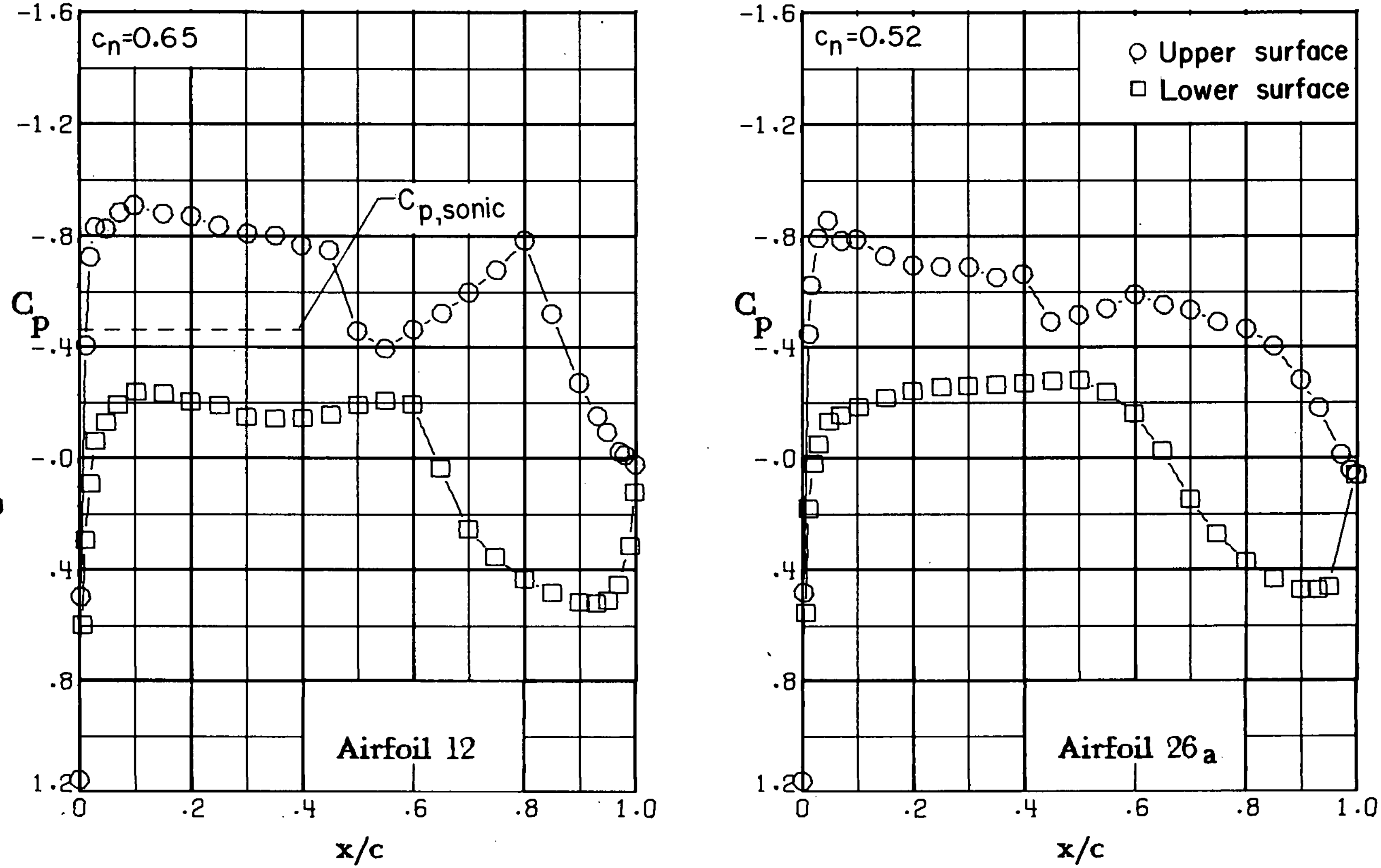
Figure 18.- Continued.



(c) M = 0.79; α = 0.5°.

Figure 18.- Continued.

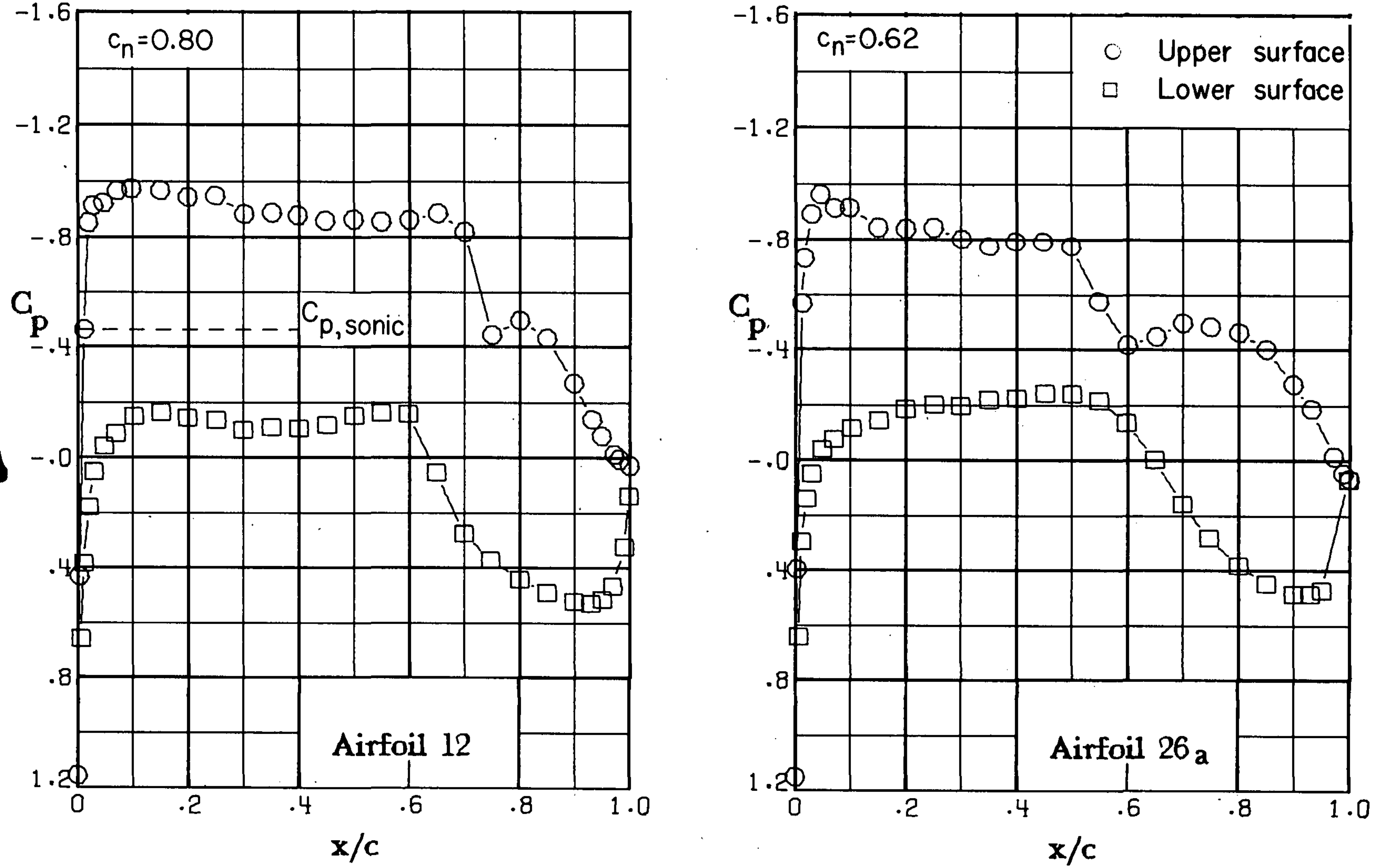
[redacted]



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

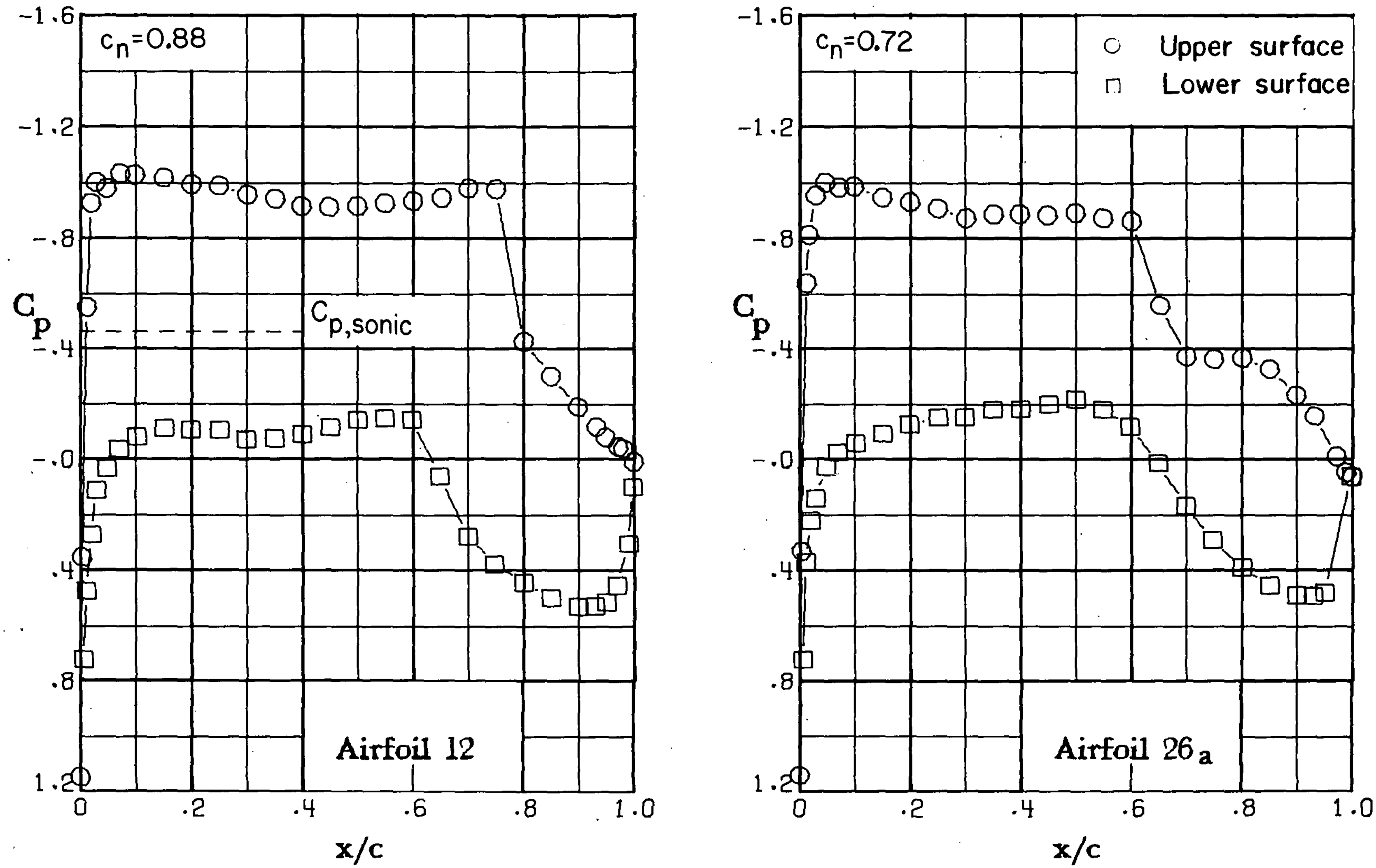
Figure 18.- Continued.

[redacted]



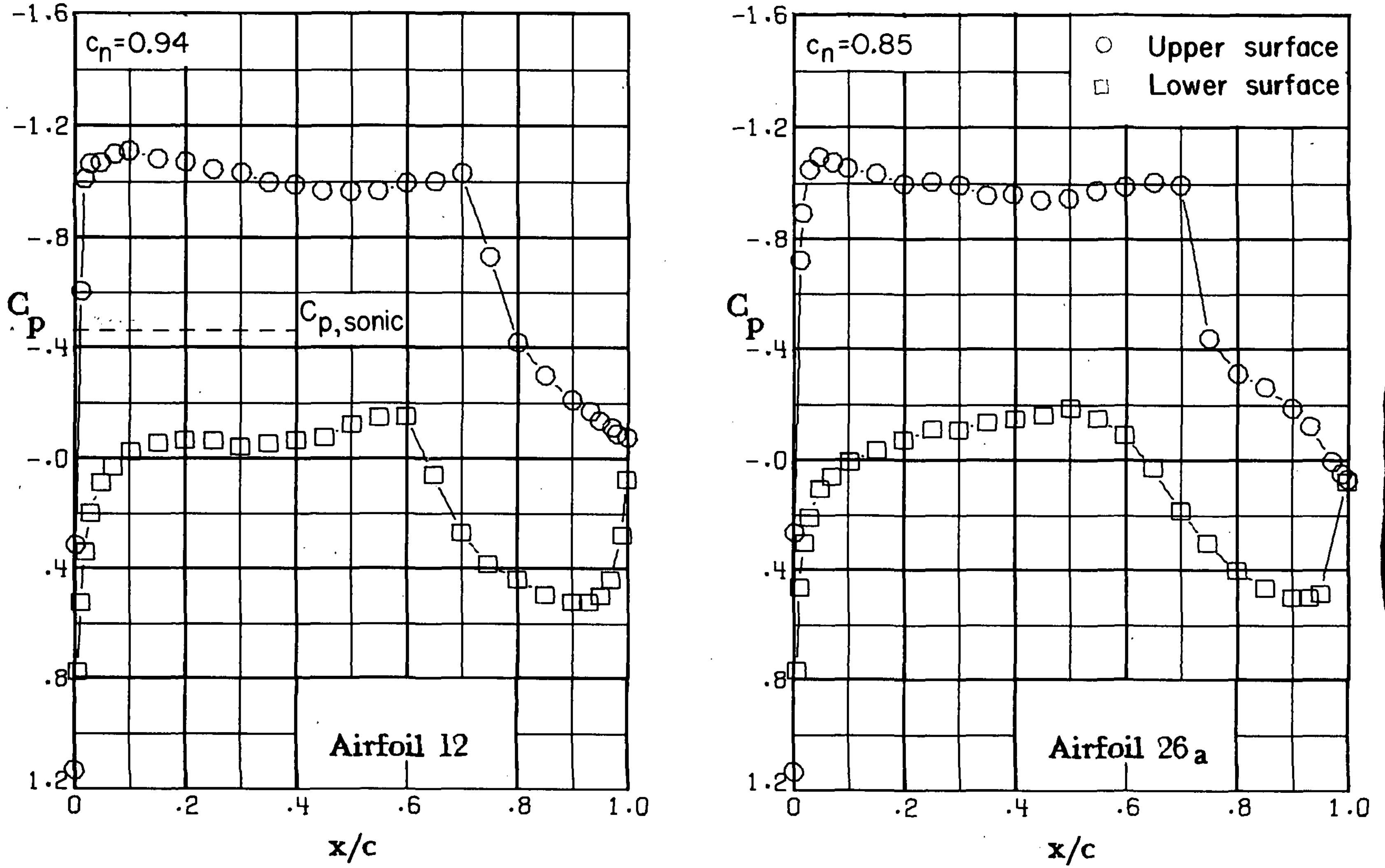
(e) $M = 0.79$; $\alpha = 1.5^{\circ}$.

Figure 18.- Continued.



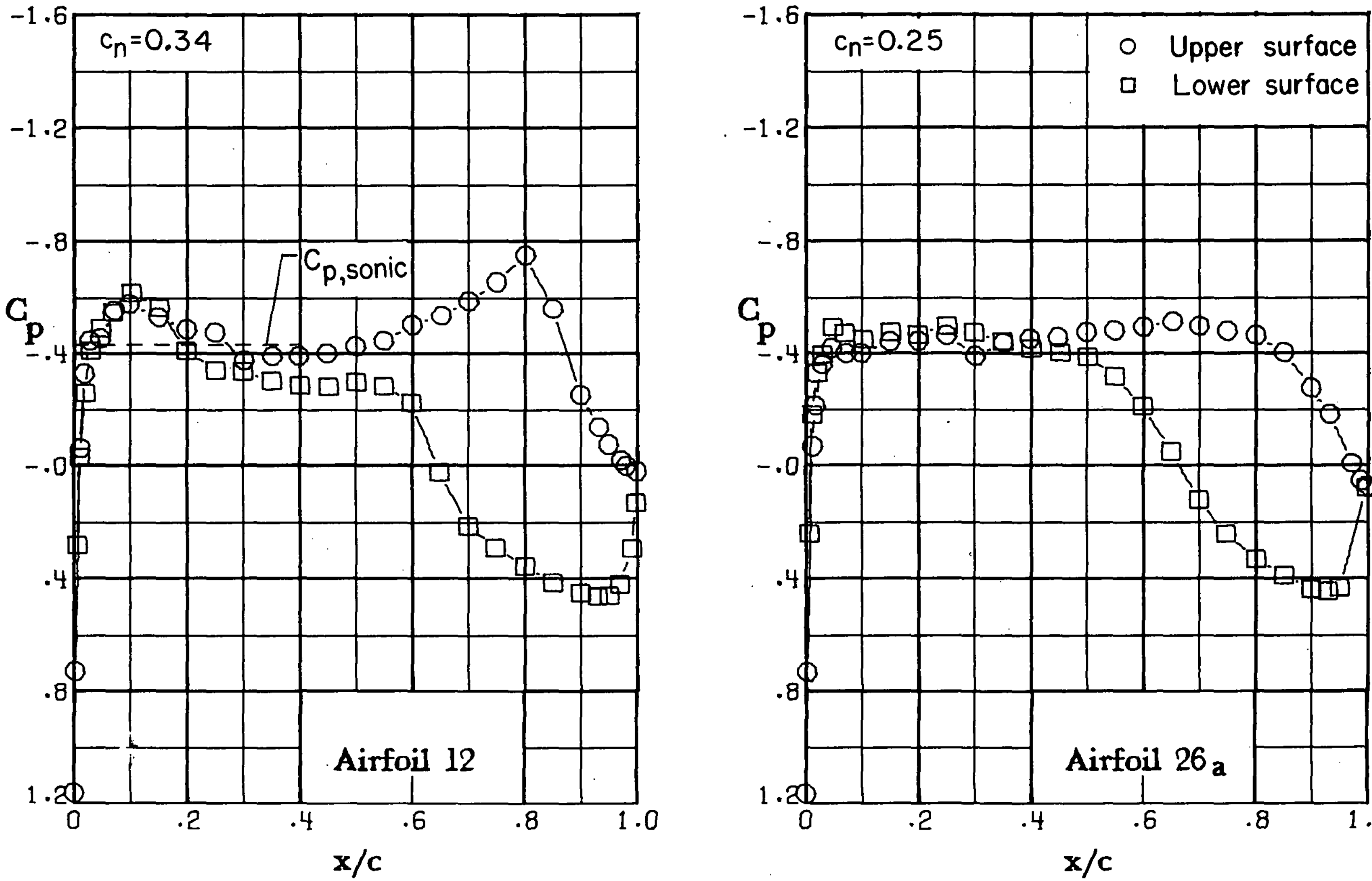
(f) M = 0.79; α = 2.0°.

Figure 18.- Continued.



(g) $M = 0.79$; $\alpha = 2.5^{o}$.

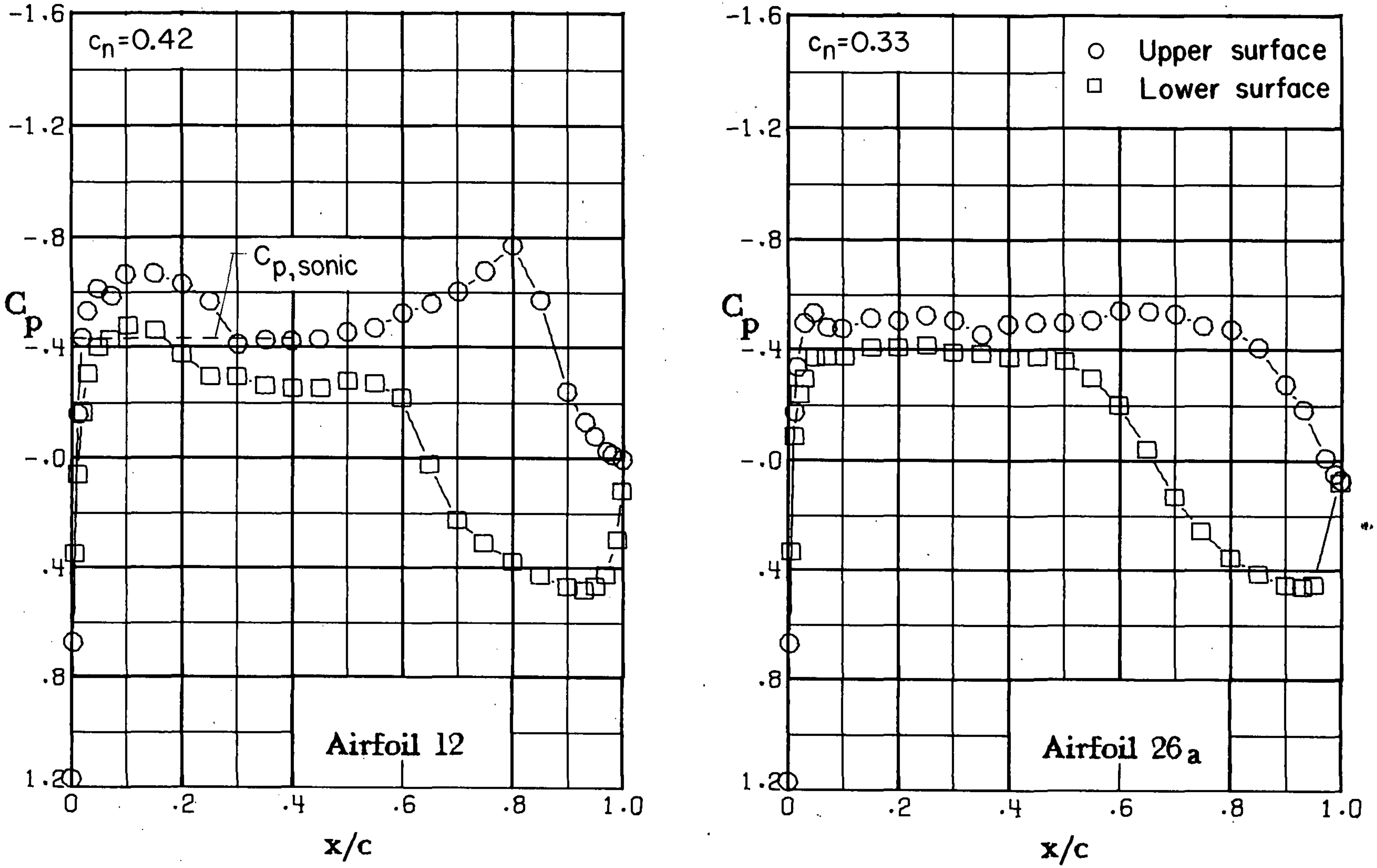
Figure 18.- Concluded.



(a) M = 0.80; α = -0.5°.

Figure 19.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.80.

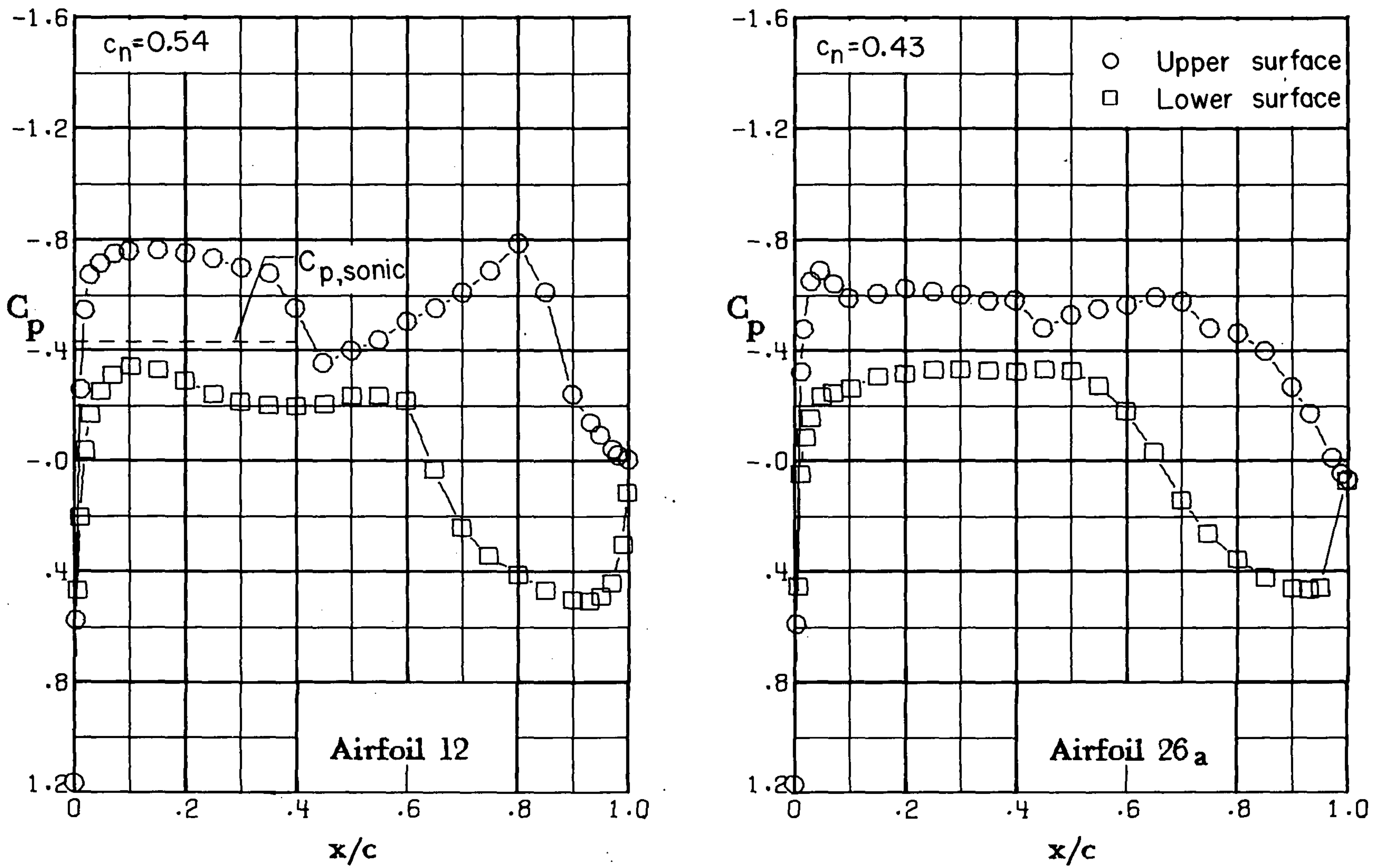
[redacted]



[redacted]

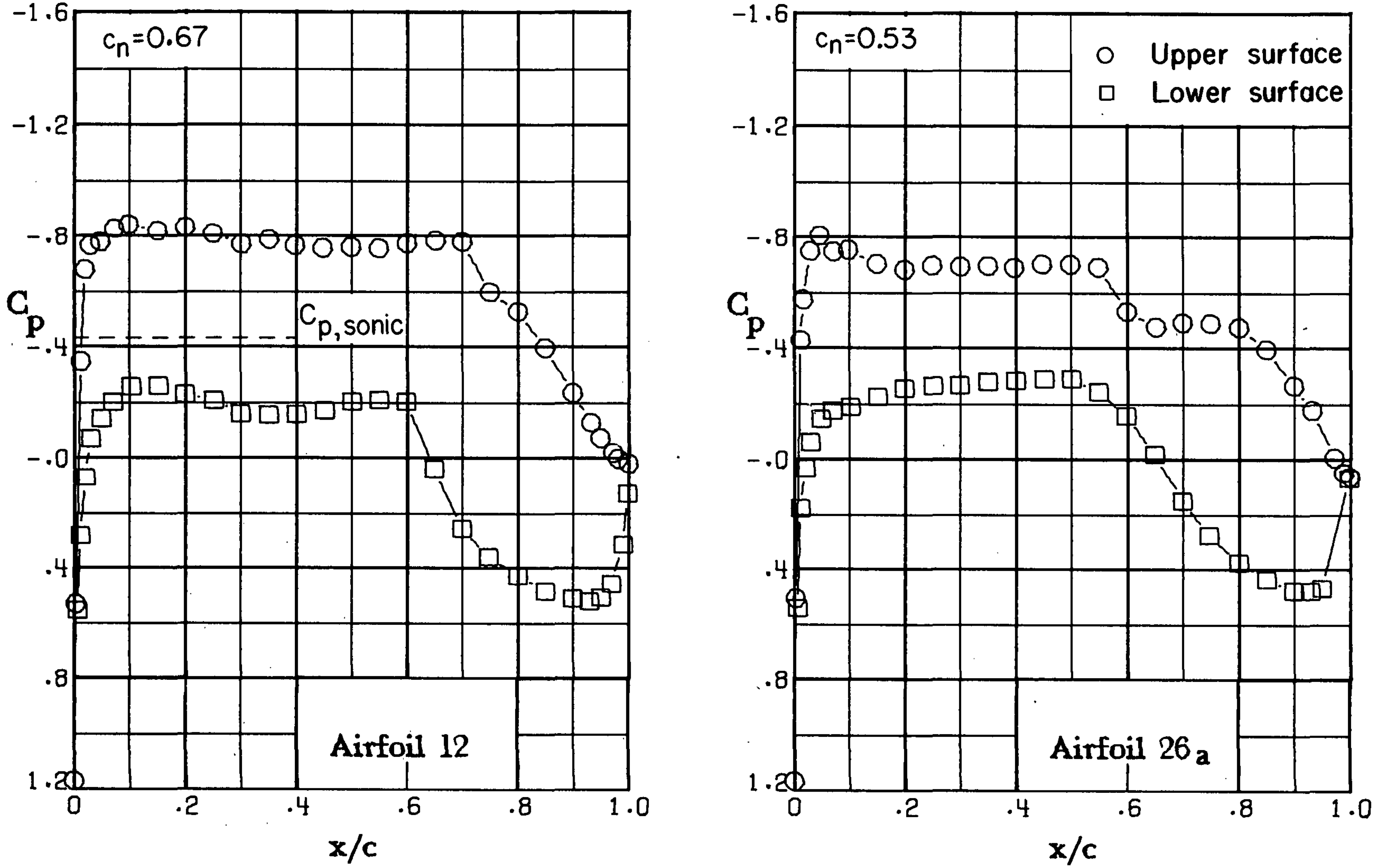
(b) M = 0.80; α = 0.0°.

Figure 19.- Continued.



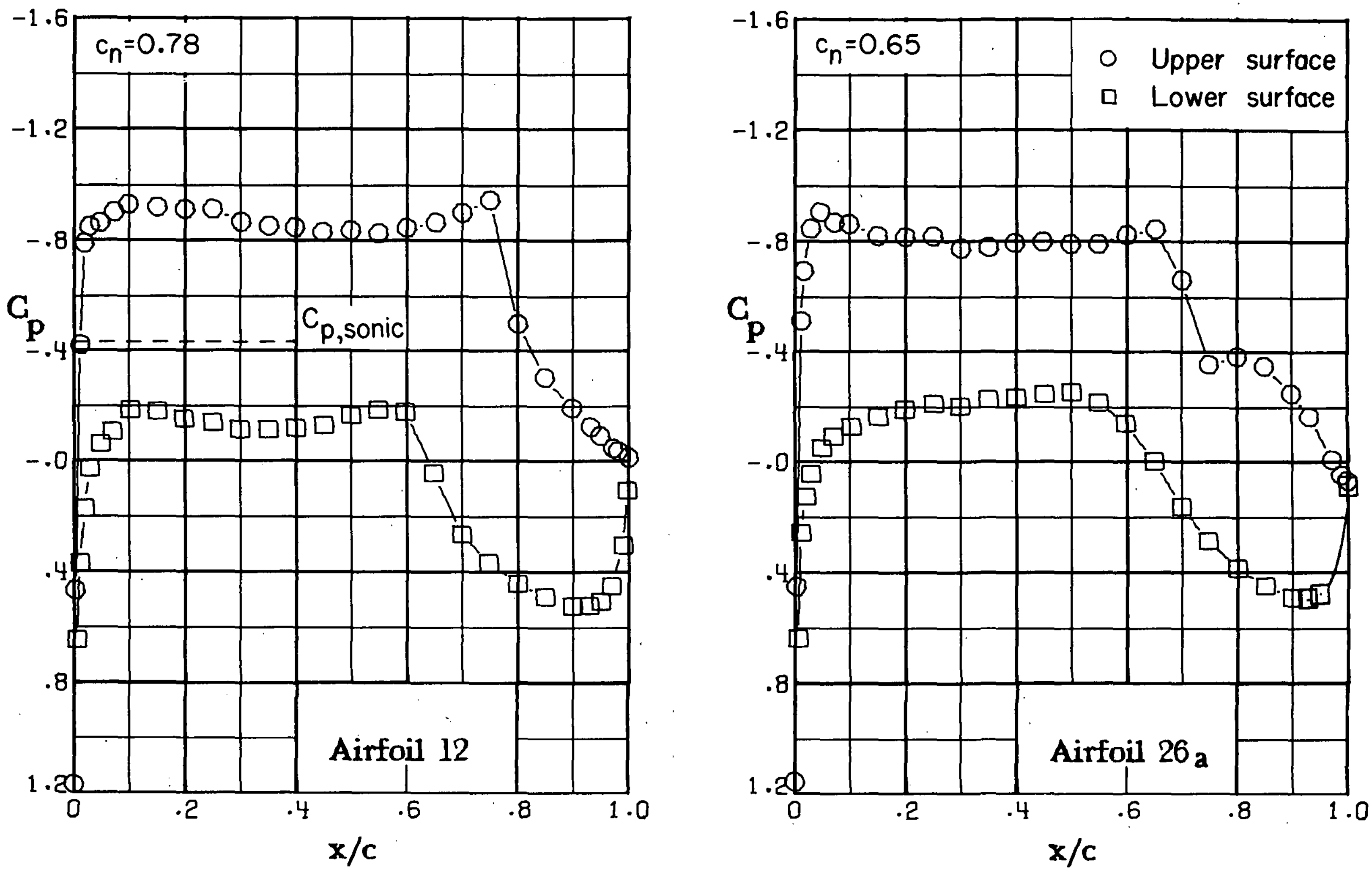
(c) M = 0.80; $\alpha = 0.5^{\circ}$.

Figure 19.- Continued.



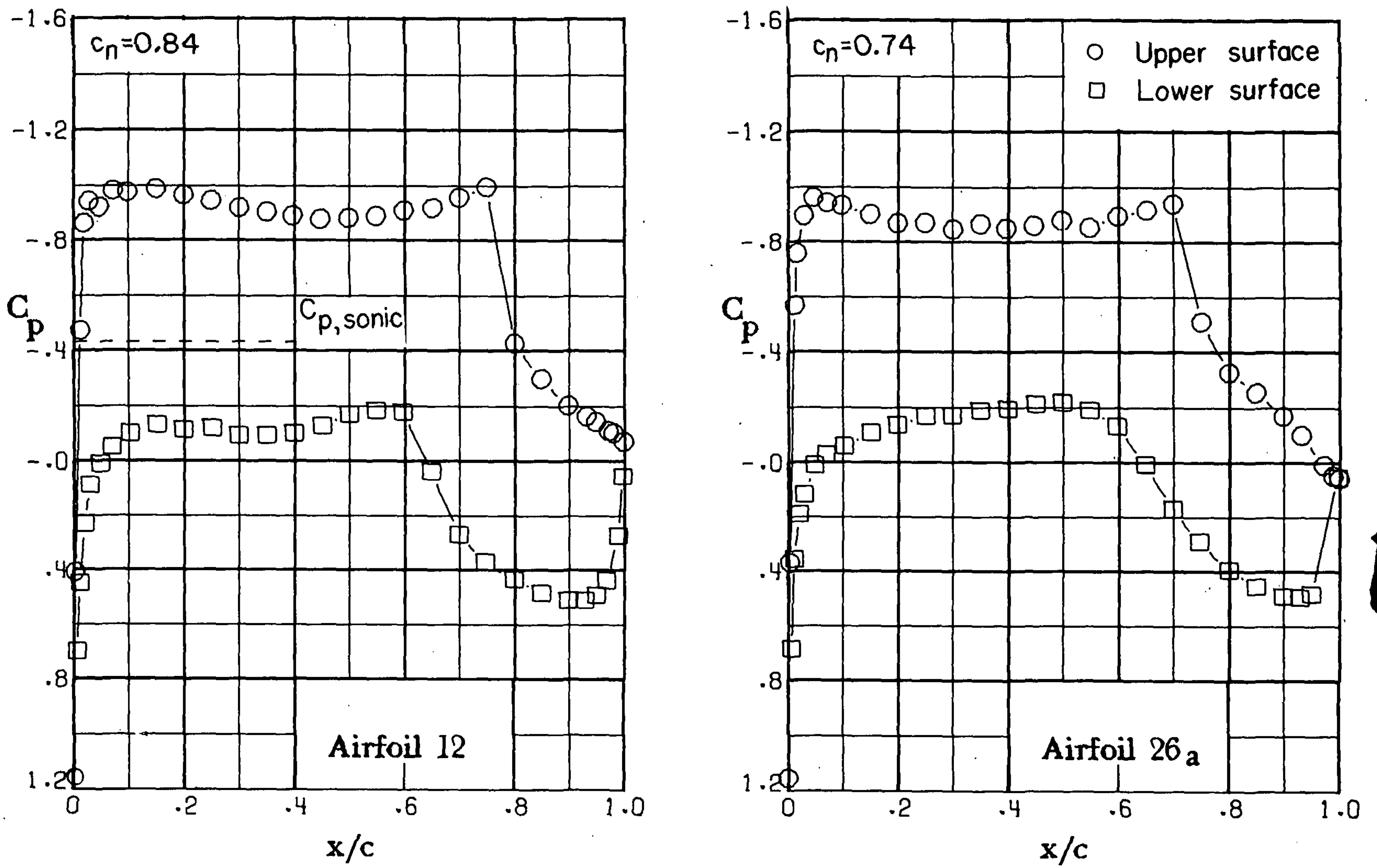
(d) M = 0.80; $\alpha = 1.0^{\circ}$.

Figure 19.- Continued.



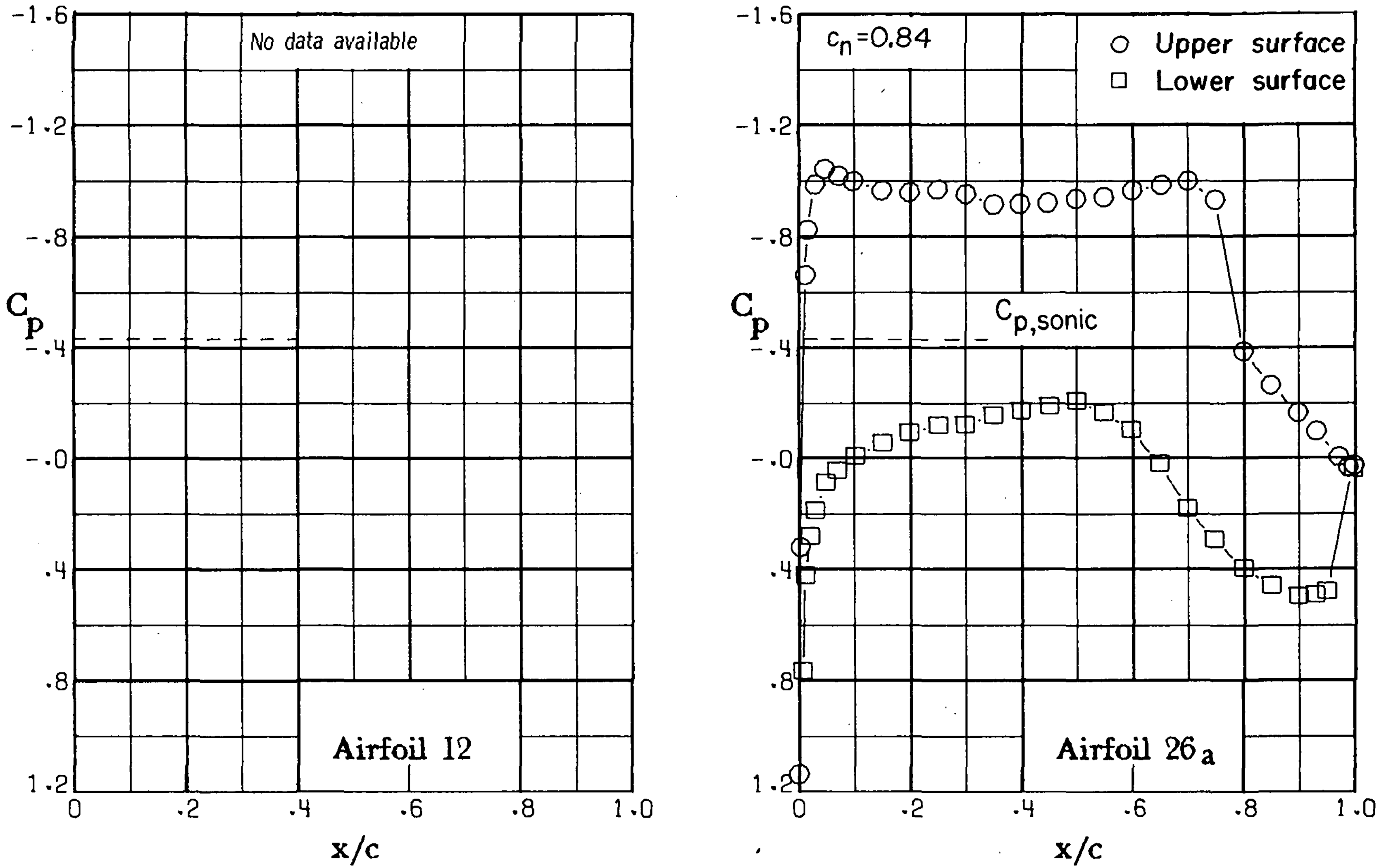
(e) M = 0.80; α = 1.5°.

Figure 19.- Continued.



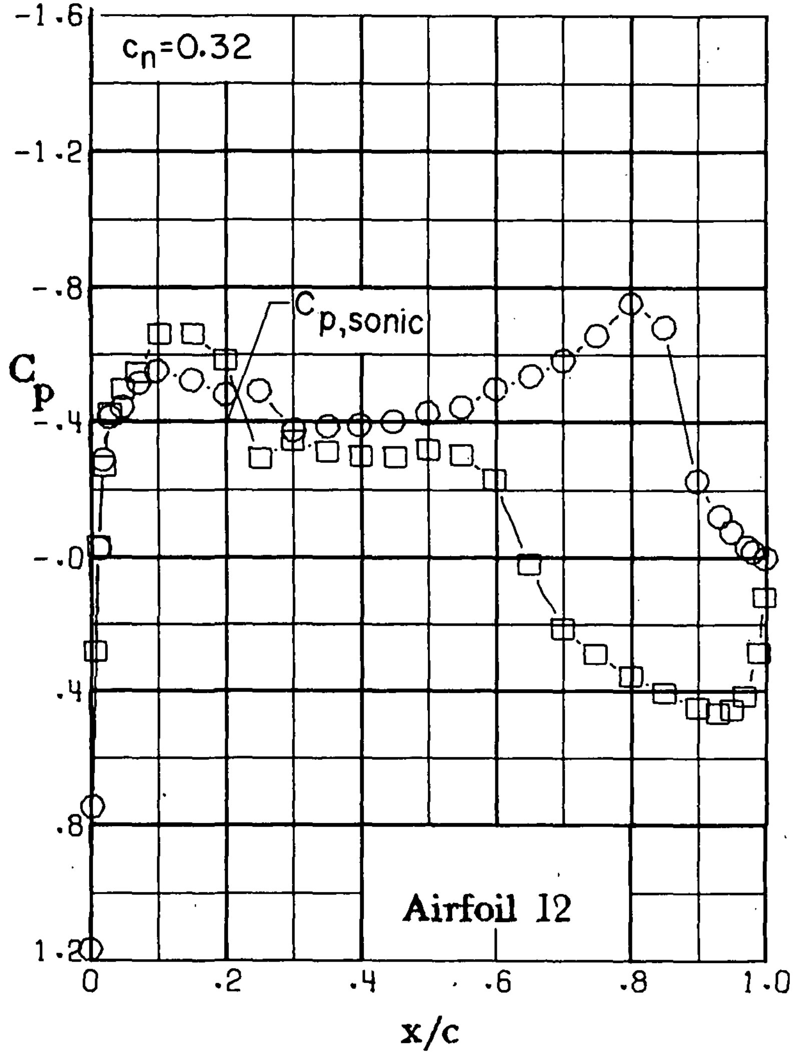
(f) M = 0.80; $\alpha = 2.0^{o}$.

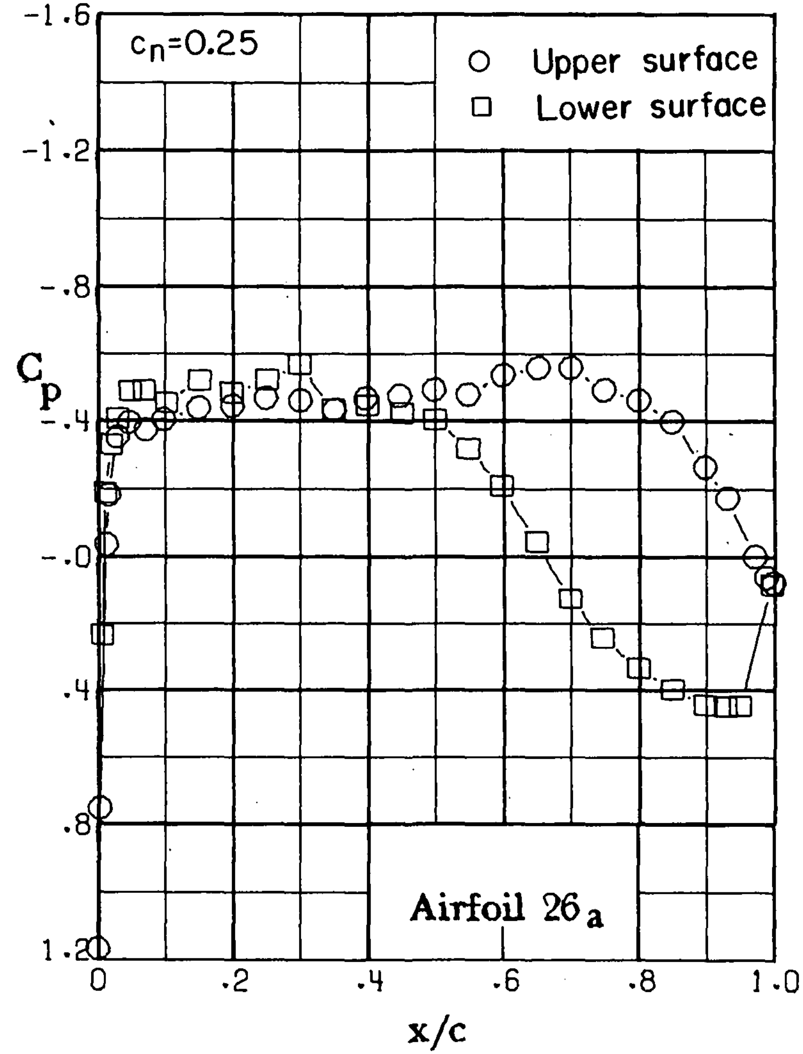
Figure 19.- Continued.



(g) $M = 0.80$; $\alpha = 2.5^{\circ}$.

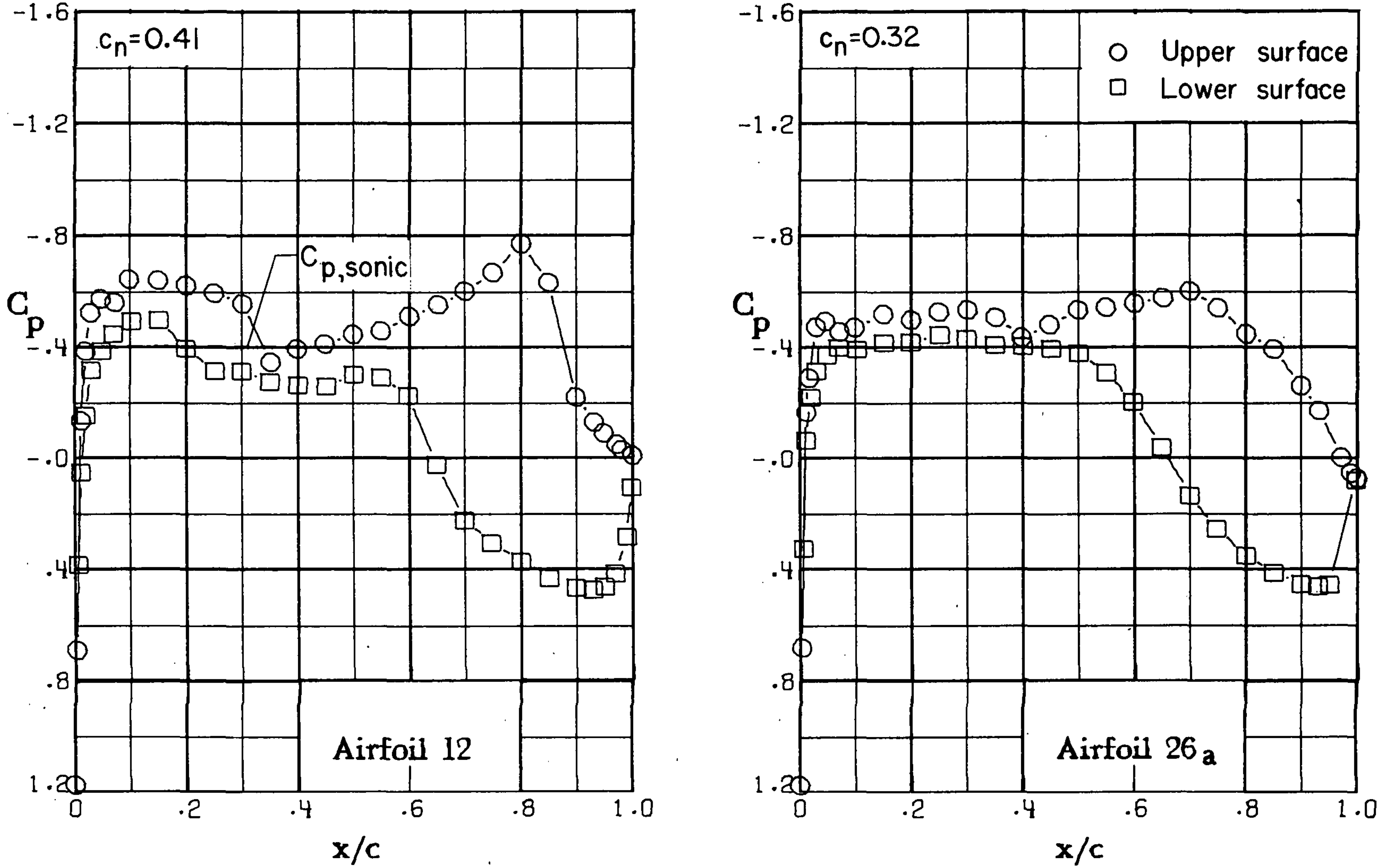
Figure 19.- Concluded.





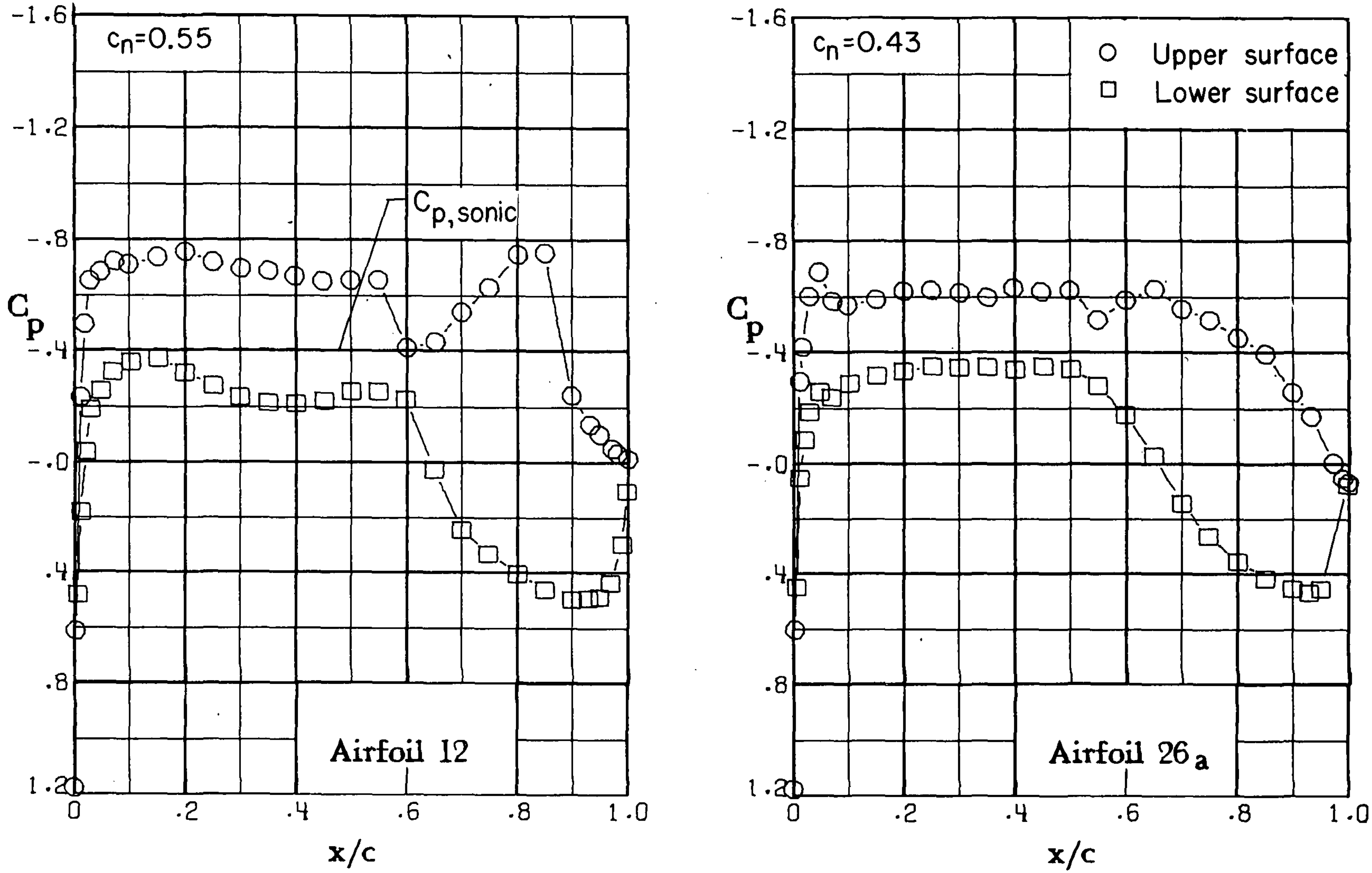
(a) M = 0.81; $\alpha \doteq -0.5^{\circ}$.

Figure 20.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.81.



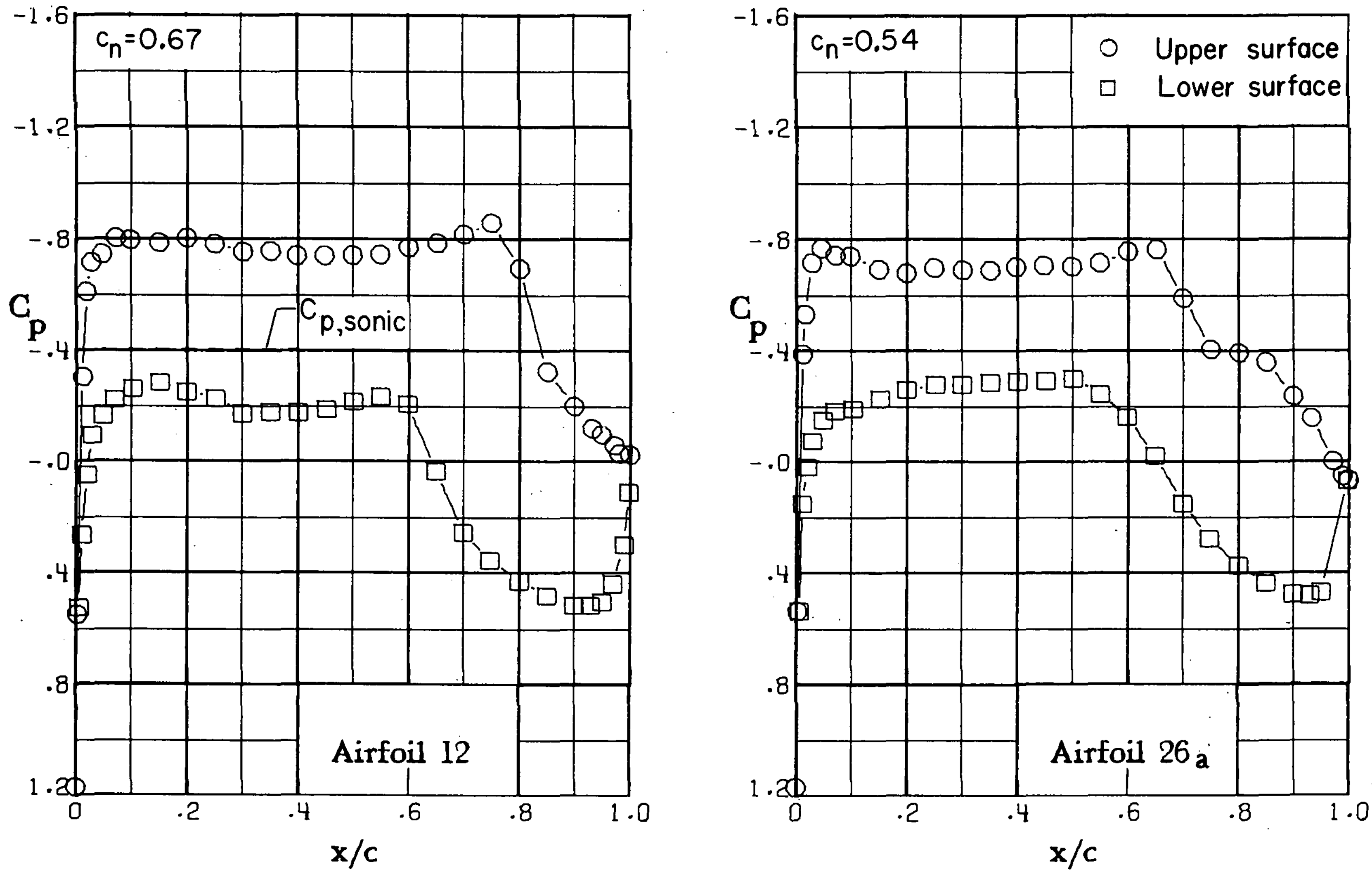
(b) $M = 0.81$; $\alpha = 0.0^{o}$.

Figure 20.- Continued.



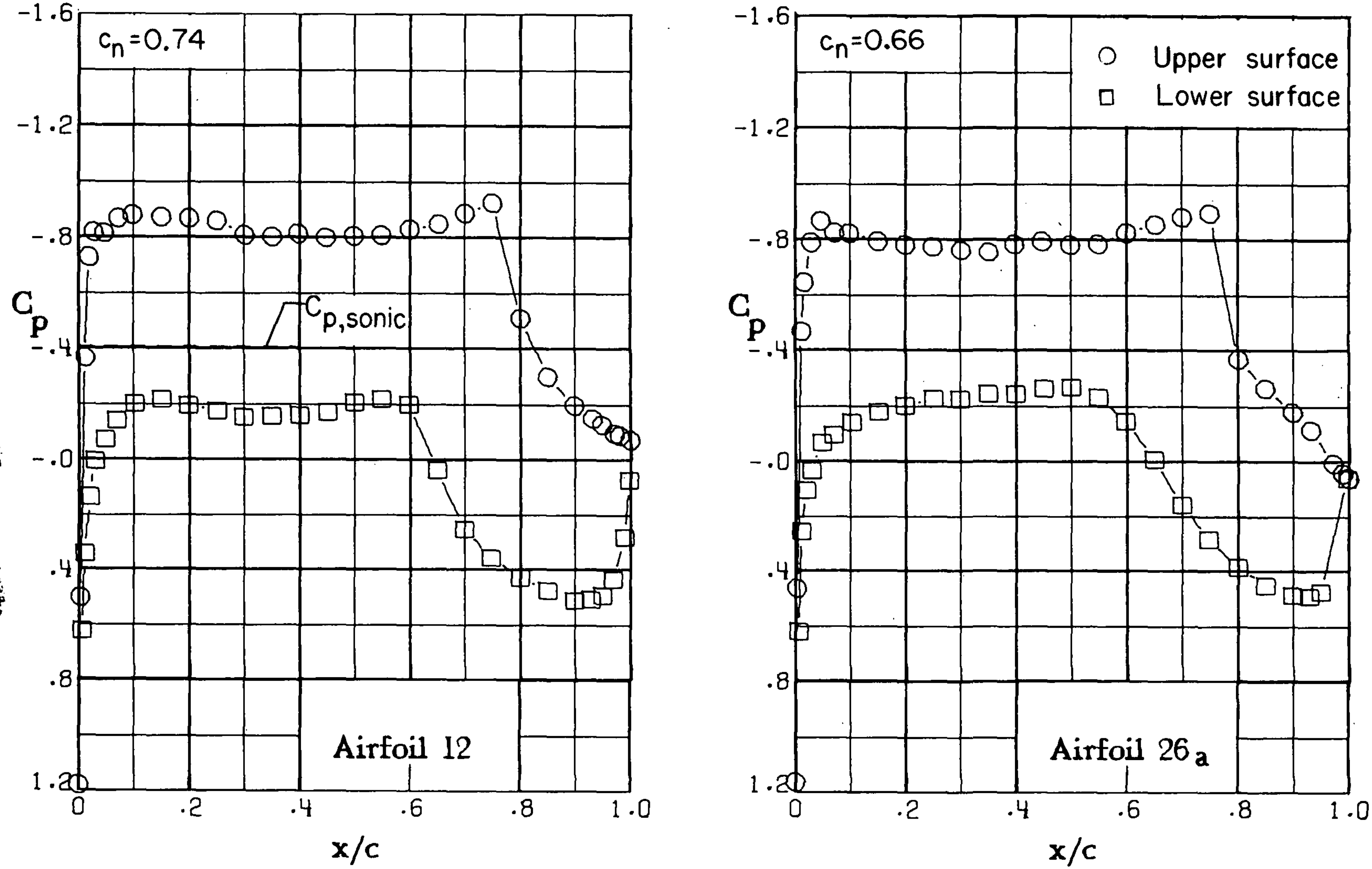
(c) M = 0.81; $\alpha = 0.5^{\circ}$.

Figure 20.- Continued.



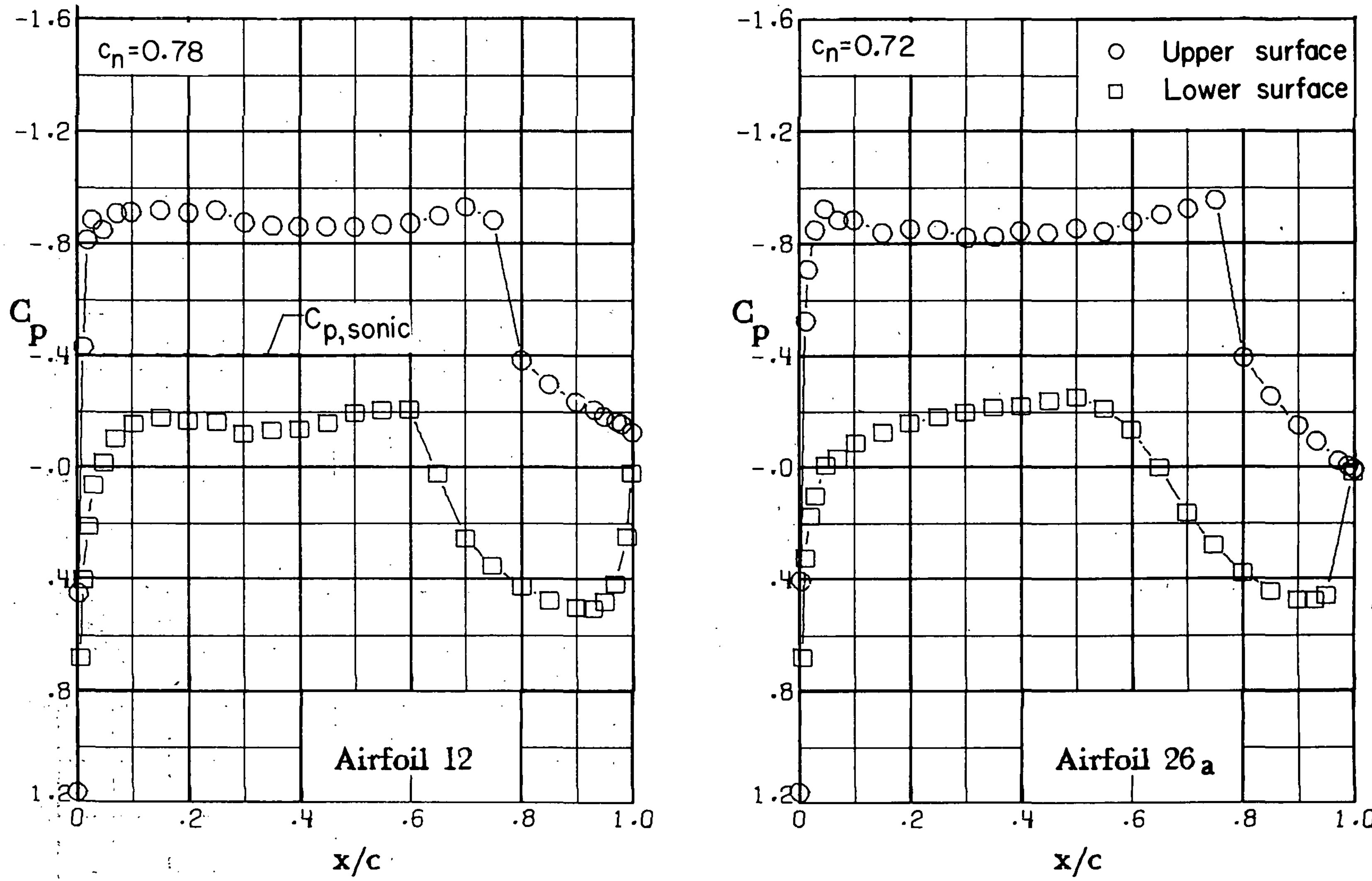
(d) M = 0.81; α = 1.0°.

Figure 20.- Continued.



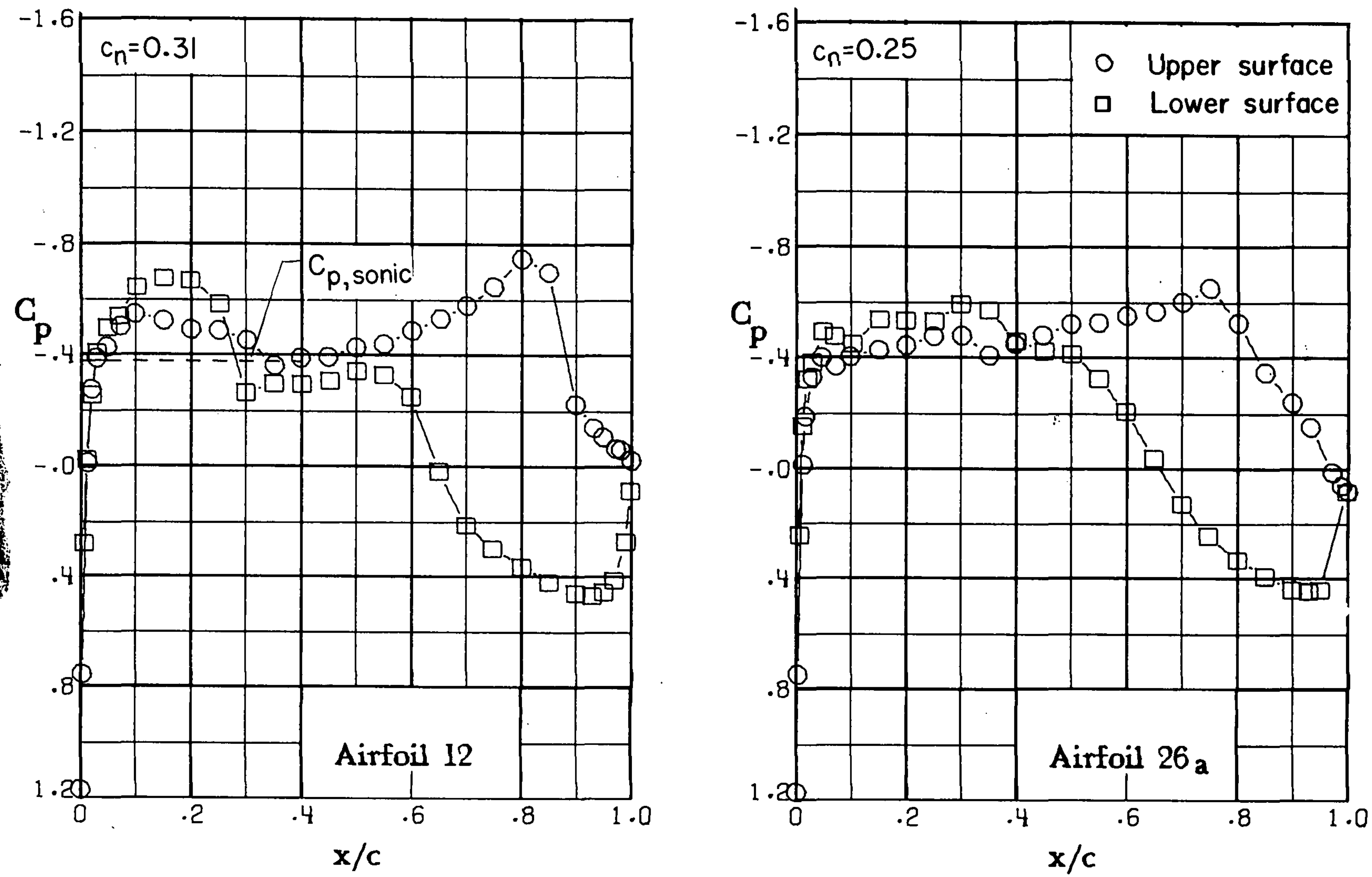
(e) M = 0.81; $\alpha = 1.5^{o}$.

Figure 20.- Continued.



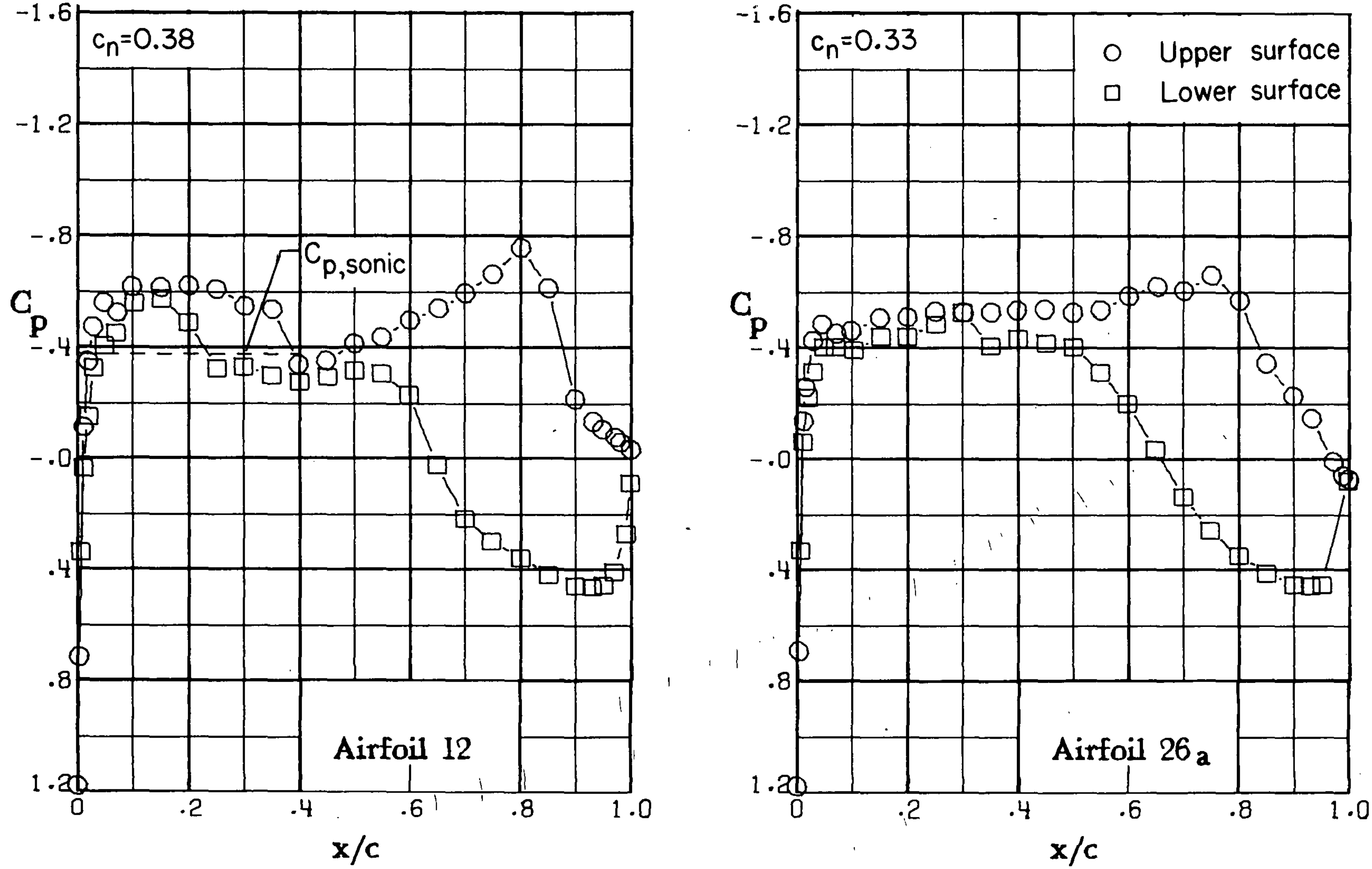
(f) M = 0.81; $\alpha = 2.0^{o}$.

Figure 20.- Concluded.



(a) M = 0.82; α = -0.5°.

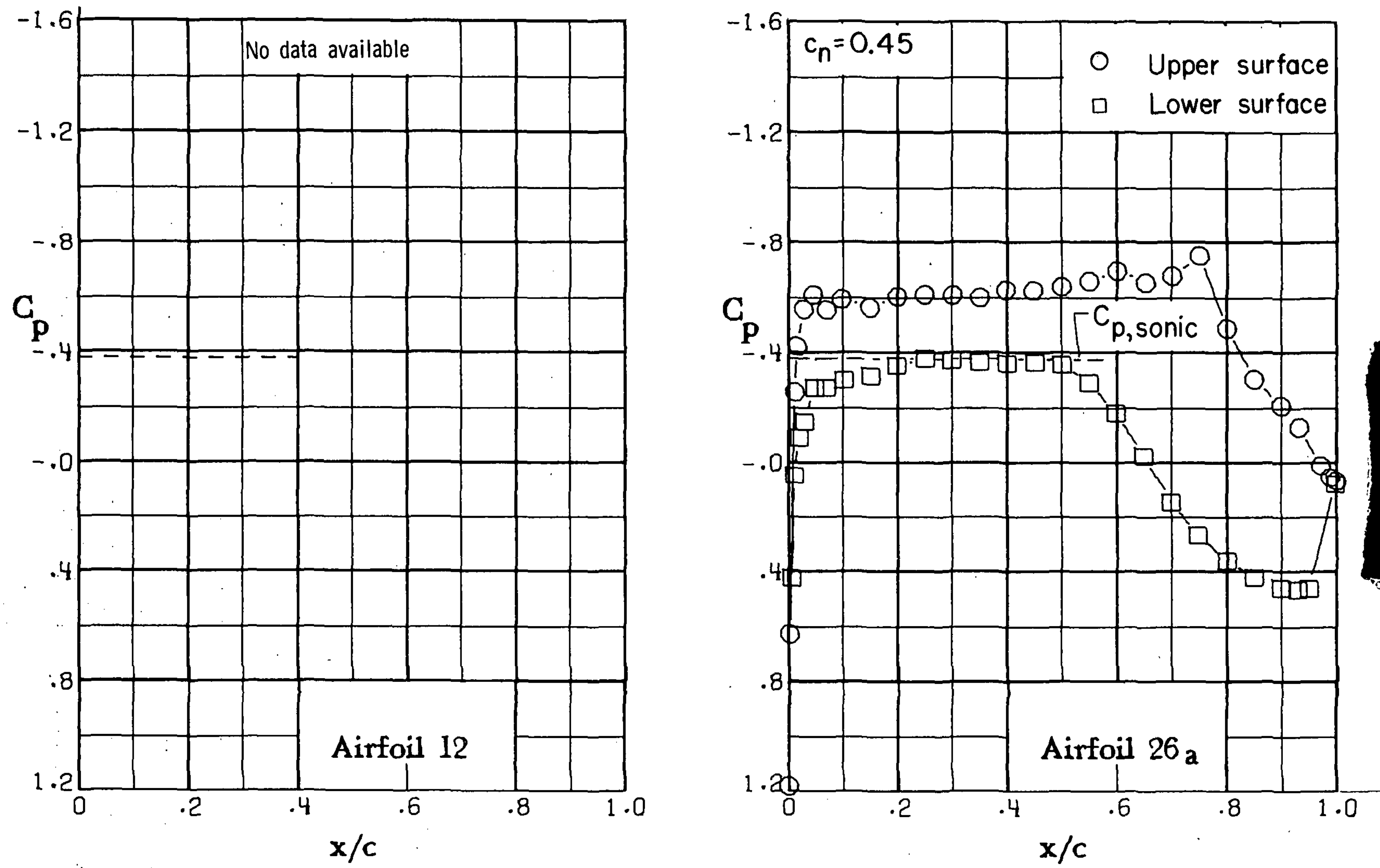
Figure 21.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.82.



(b) $M = 0.82$; $\alpha = 0.0^{\circ}$.

Figure 21.- Continued.

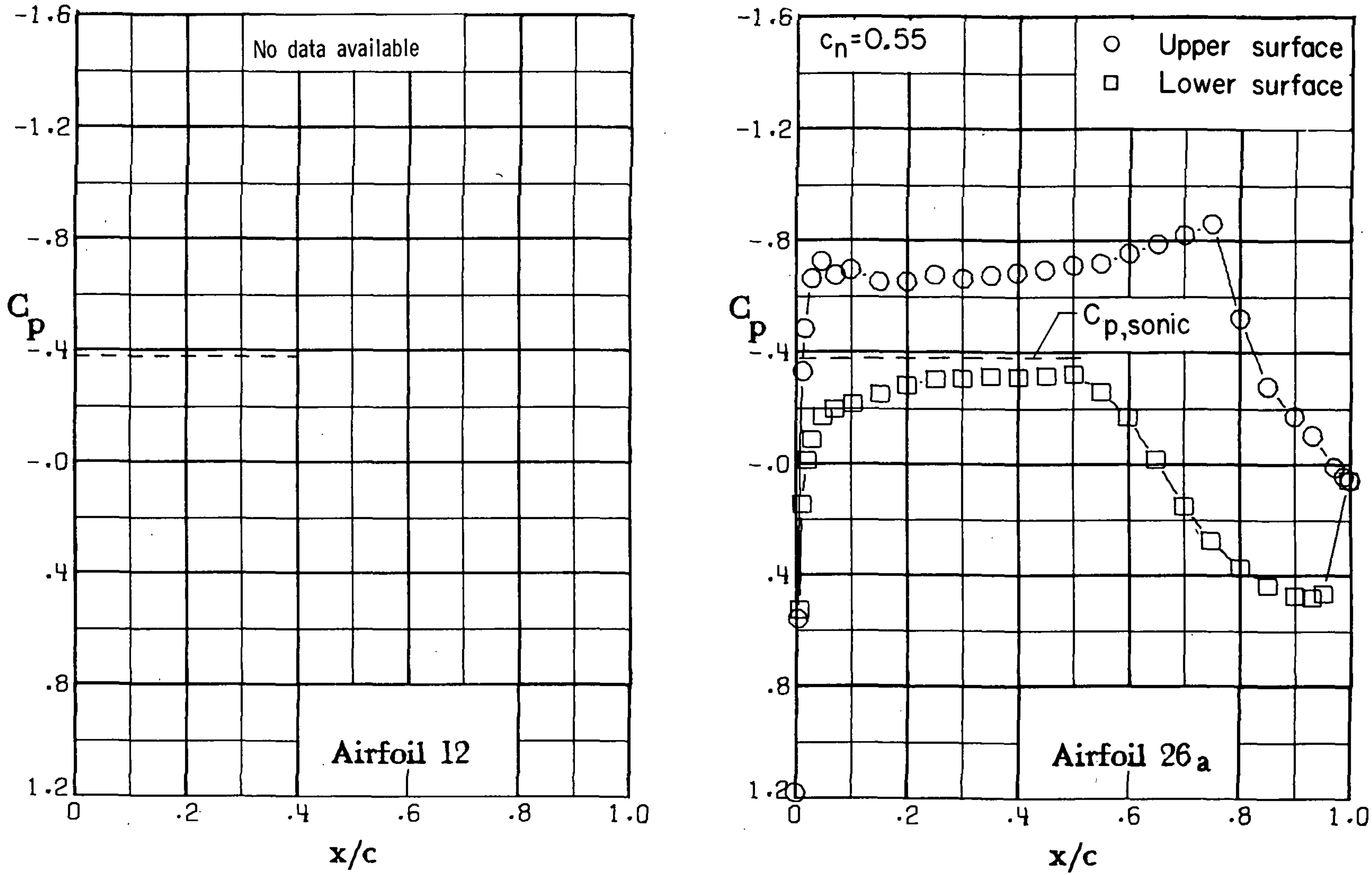
[redacted]



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

Figure 21.- Continued.

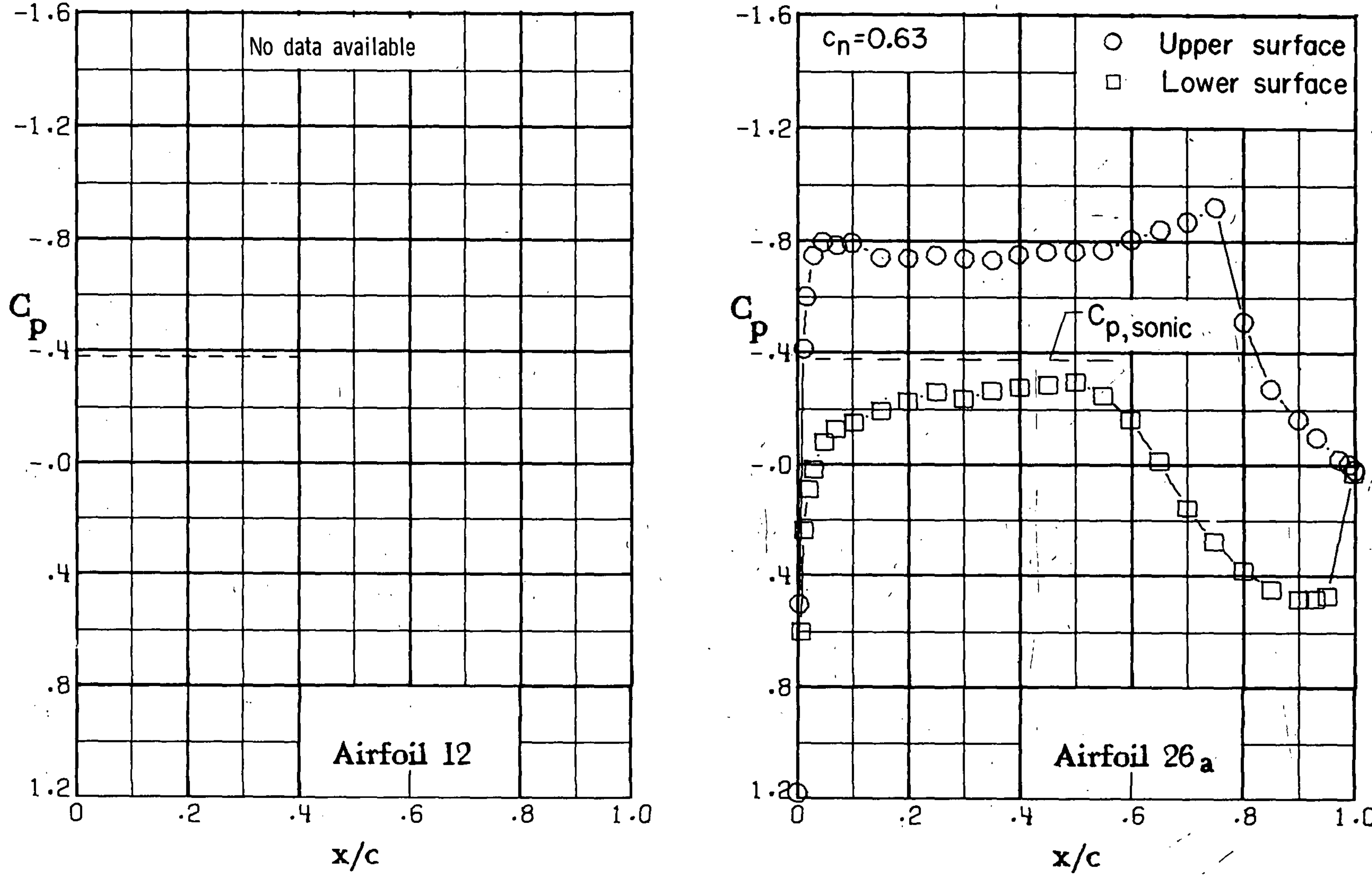
[redacted]



(d) $M = 0.82$; $\alpha = 1.0^{\circ}$.

Figure 21.- Continued.

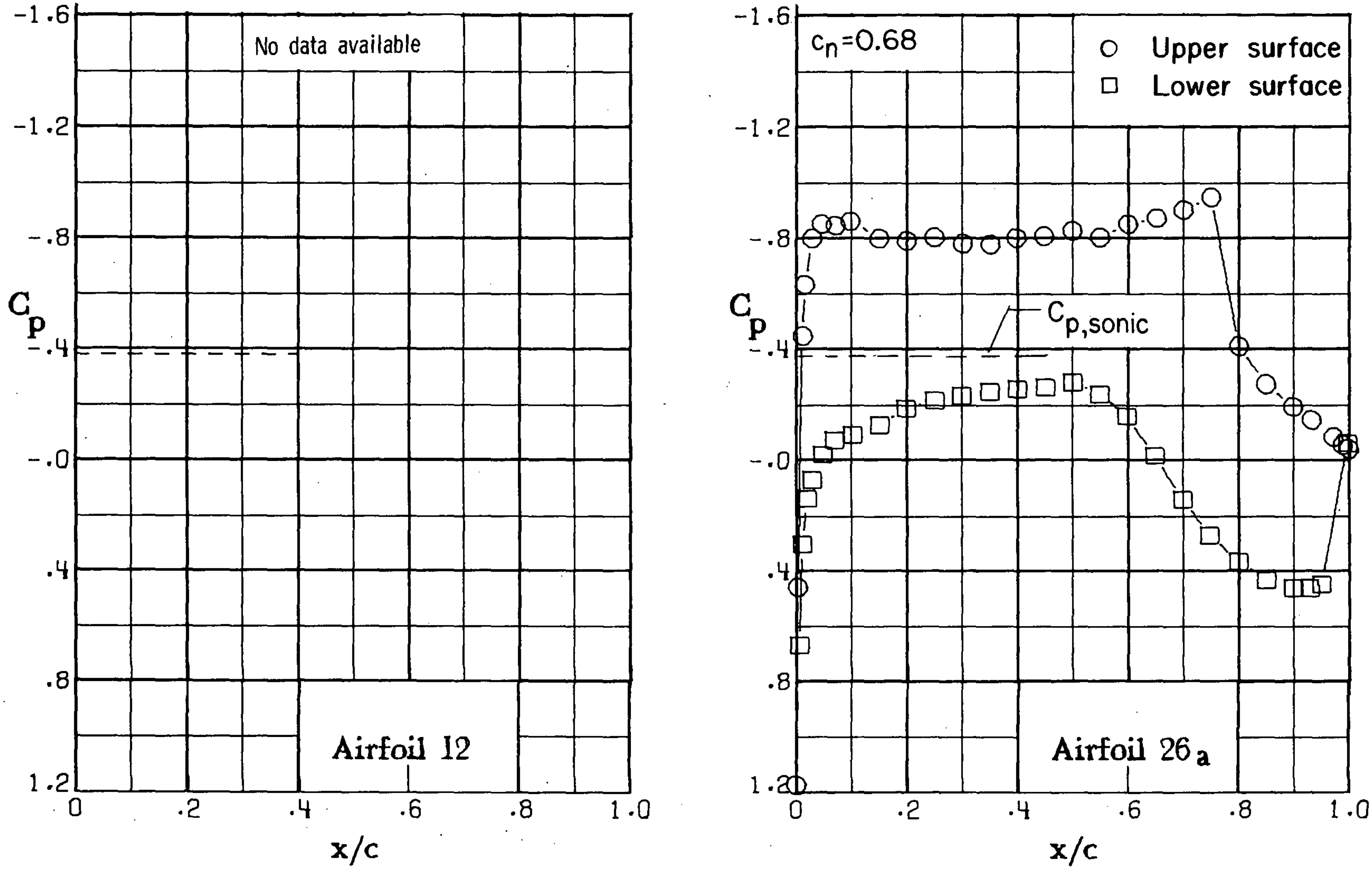
[redacted]



(e) $M = 0.82$; $\alpha = 1.5^{o}$.

Figure 21.- Continued.

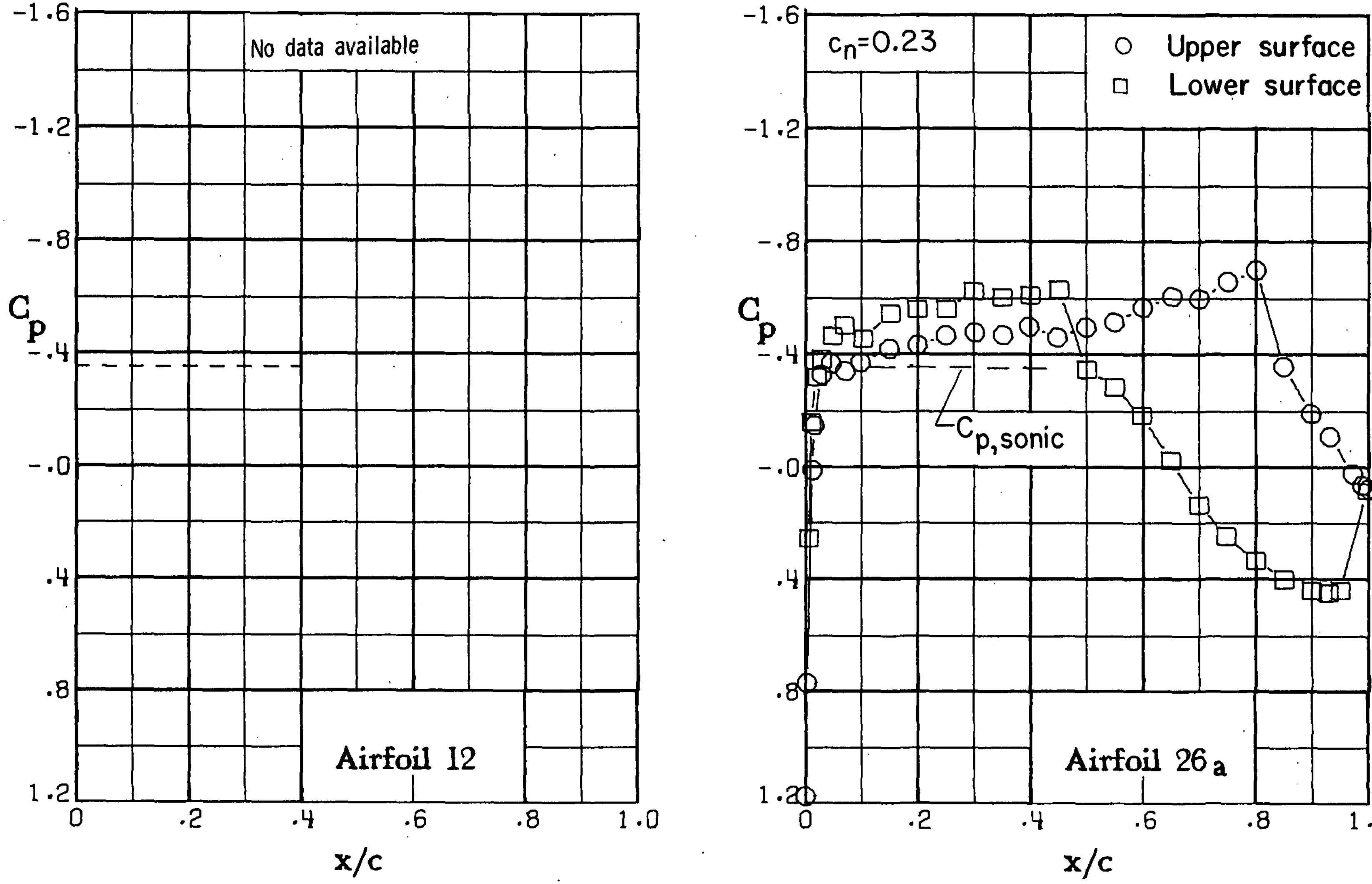
[redacted]



(f) $M = 0.82$; $\alpha = 2.0^{\circ}$.

Figure 21.- Concluded.

[redacted]

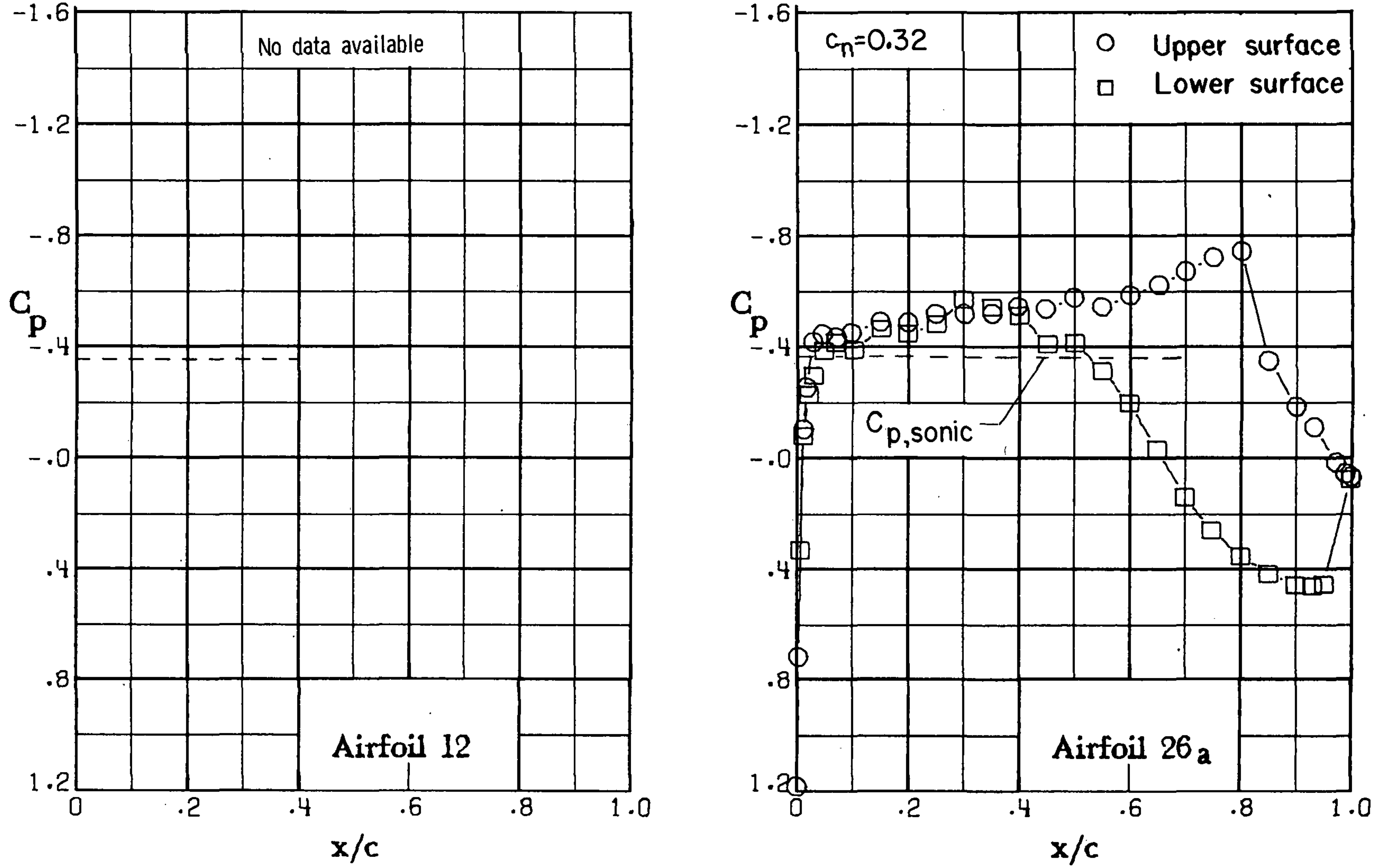


(a) M = 0.83; α = -0.5°.

Figure 22.- Chordwise pressure distributions for supercritical airfoils 12 and 26a. M = 0.83.

[redacted]

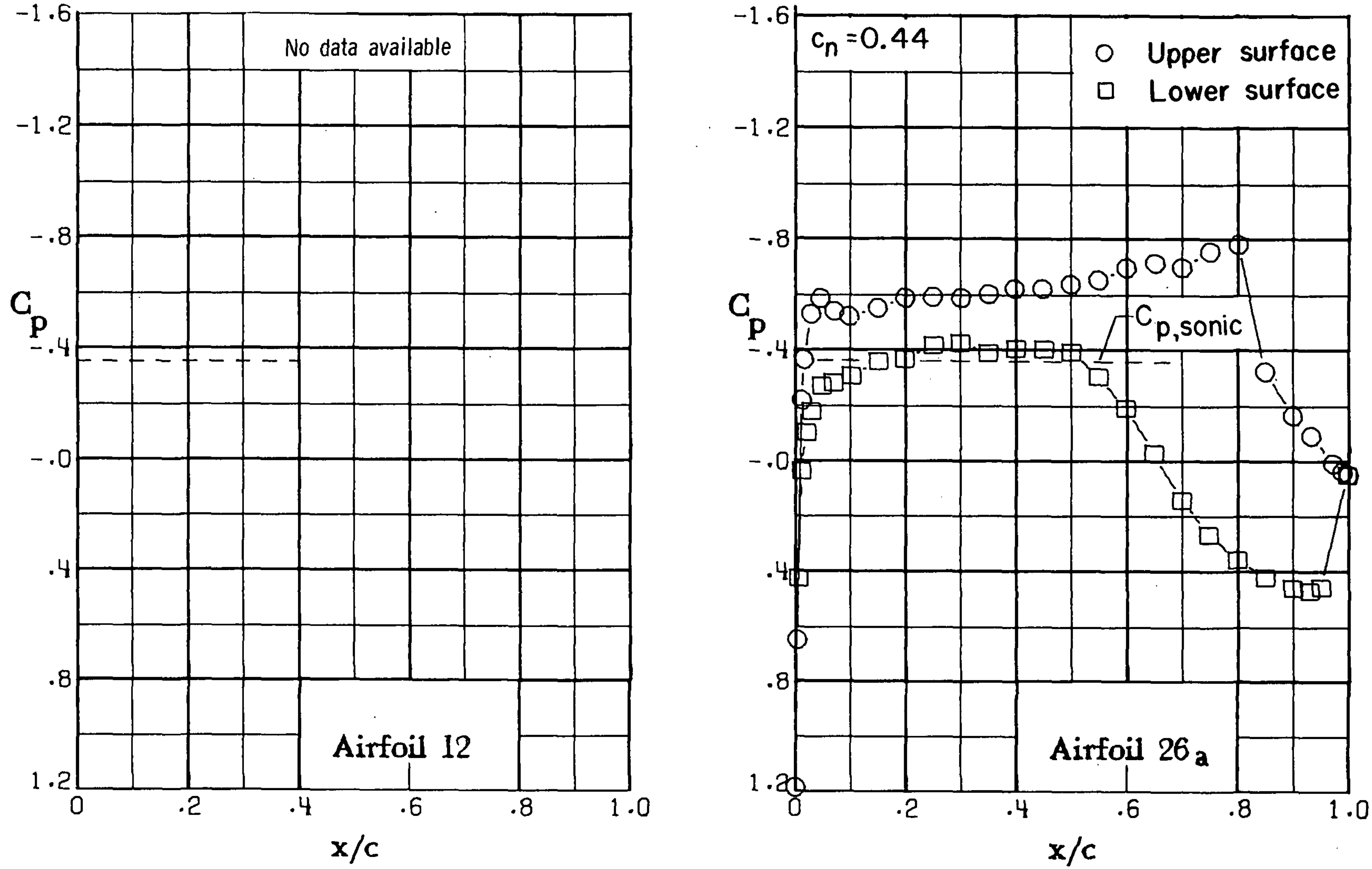
[redacted]



[redacted]

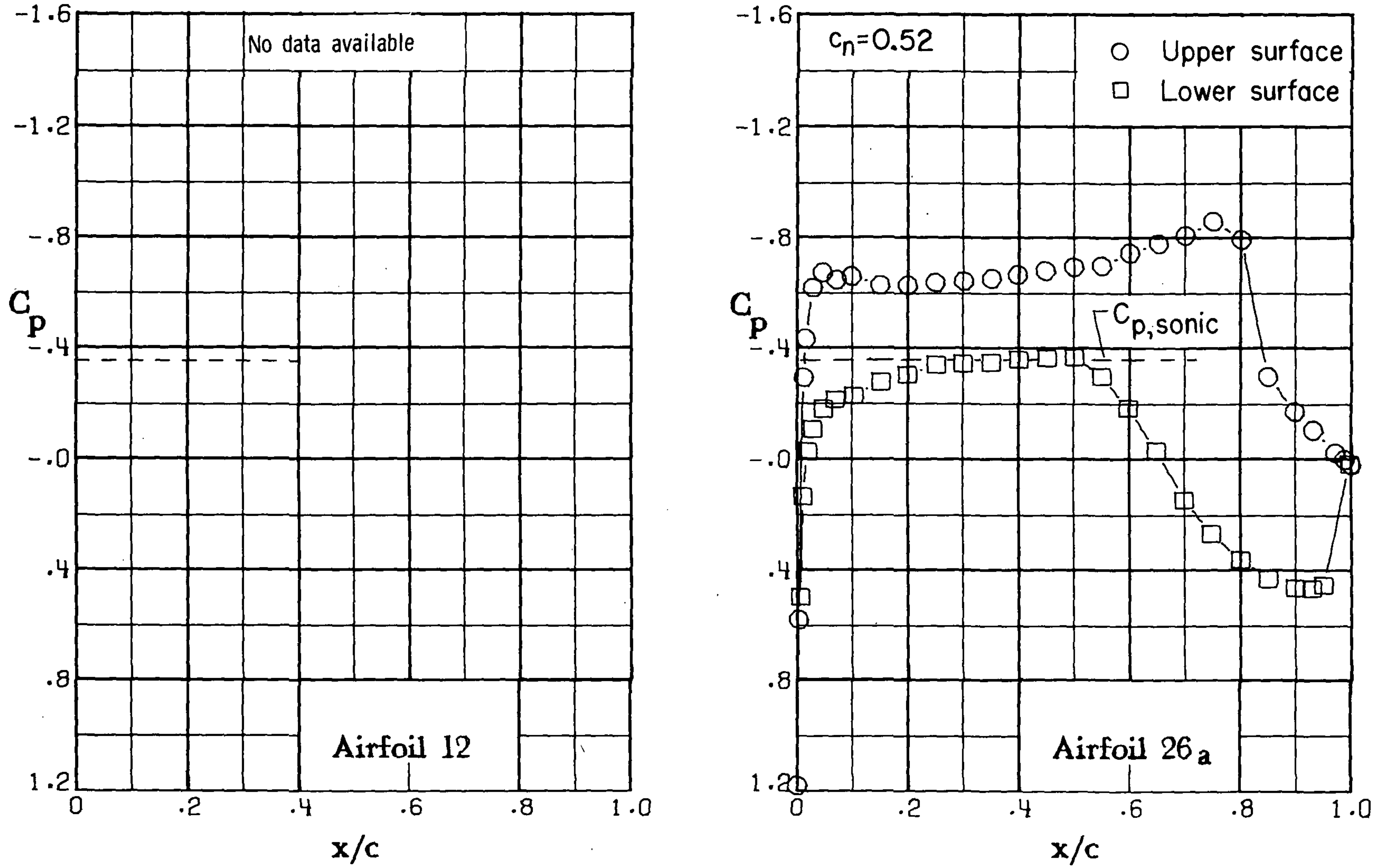
(b) M = 0.83; $\alpha = 0.0^{o}$.

Figure 22.- Continued.



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

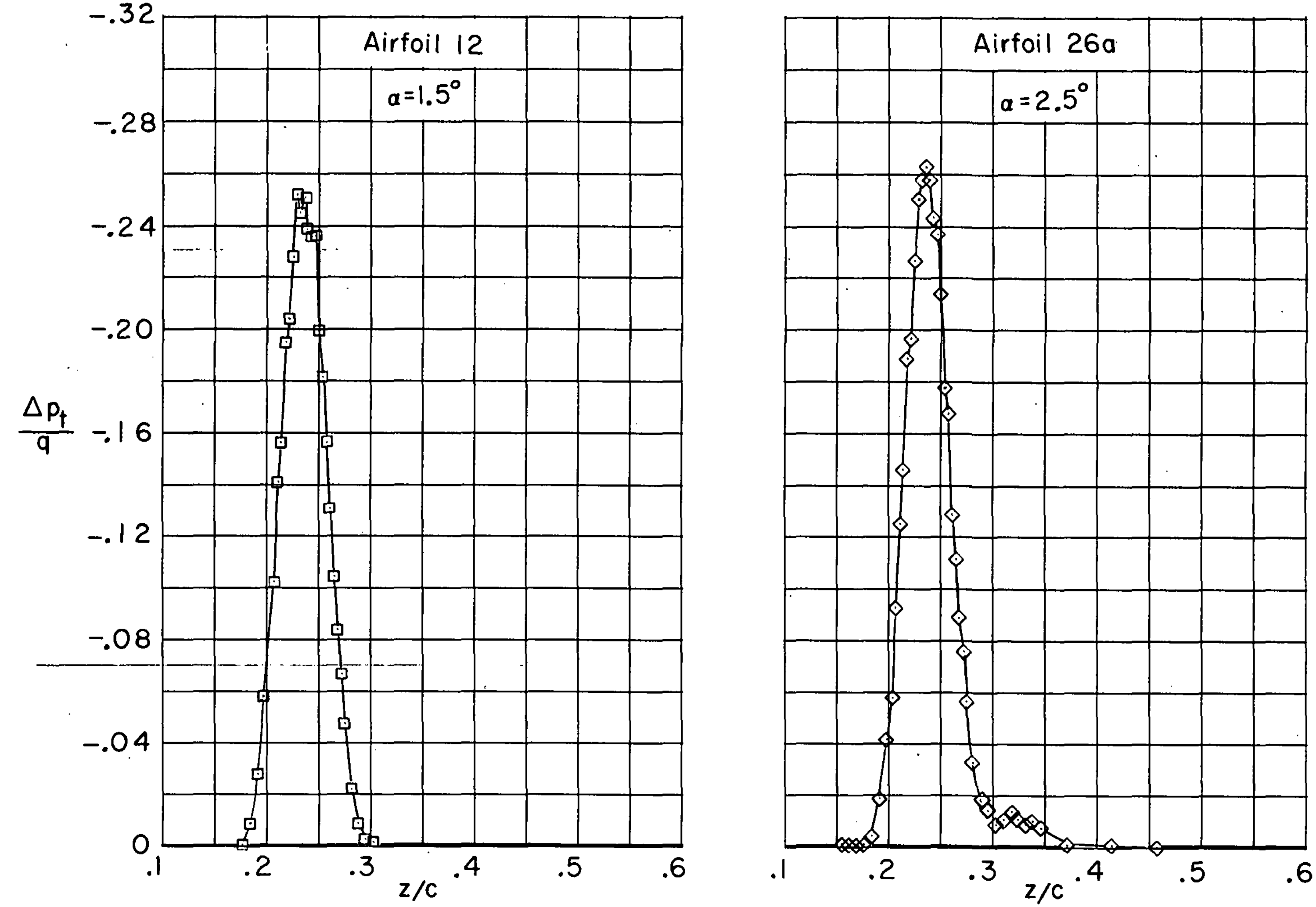
Figure 22.- Continued.



(d) M = 0.83; $\alpha = 1.0^{o}$.

Figure 22.- Concluded.

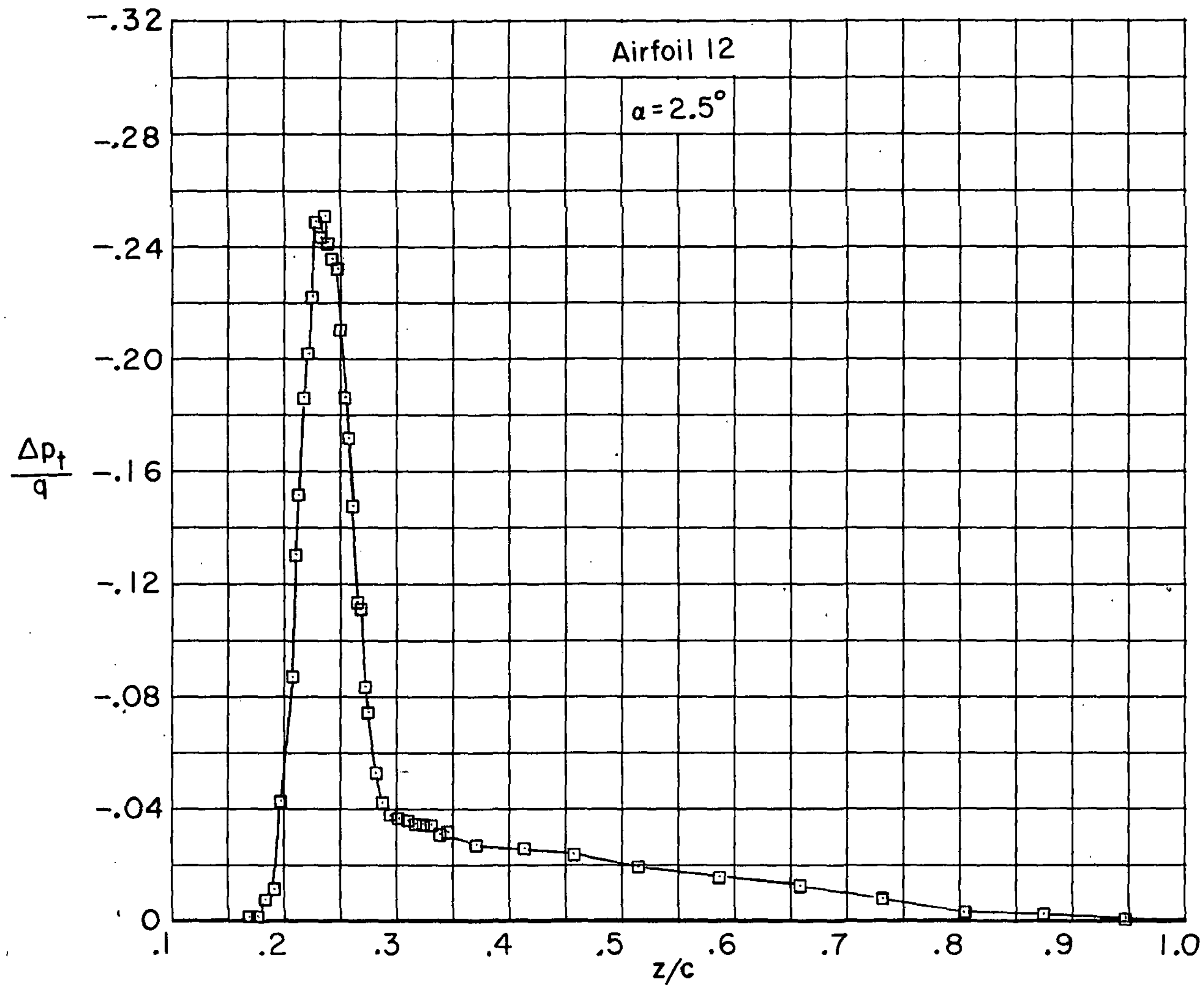
[redacted]



(a) M = 0.70; $c_n \approx 0.66$.

Figure 23.- Representative wake profiles.

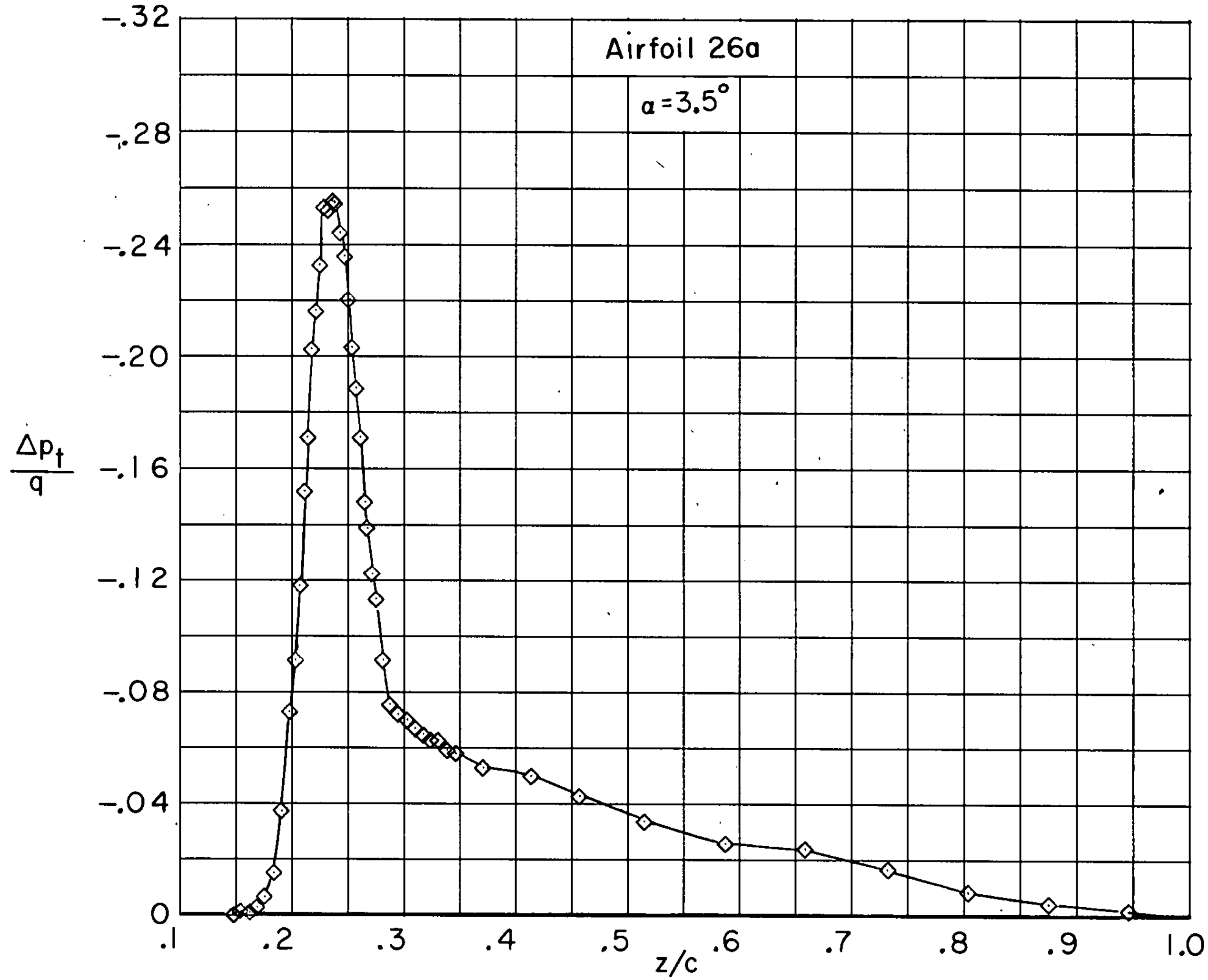
[redacted]



(b) $M = 0.76$; $c_n \approx 0.96$.

Figure 23.- Continued.

[redacted]

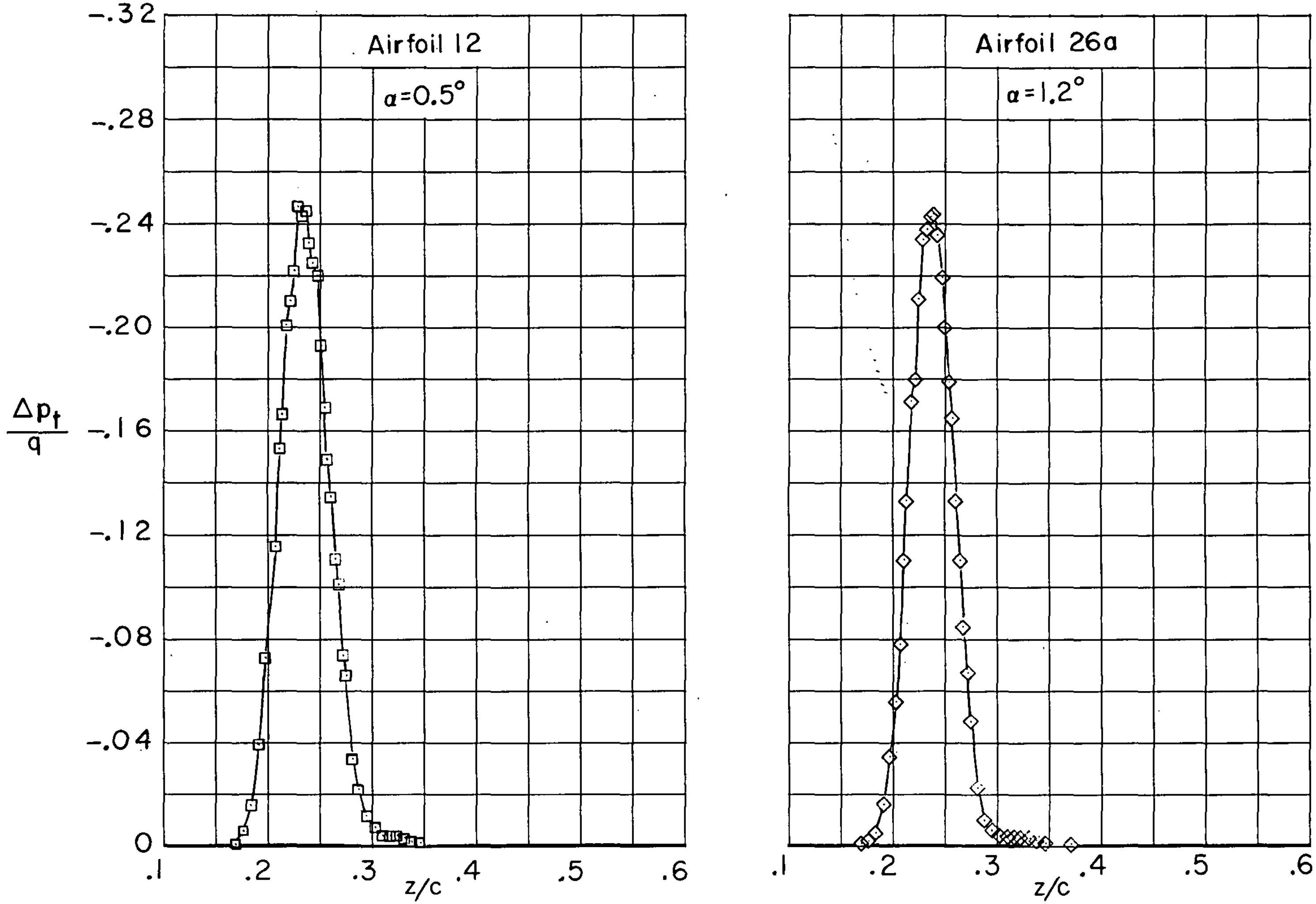


(b) M = 0.76; $c_n \approx 0.96$. Concluded.

Figure 23.- Continued.

[redacted]

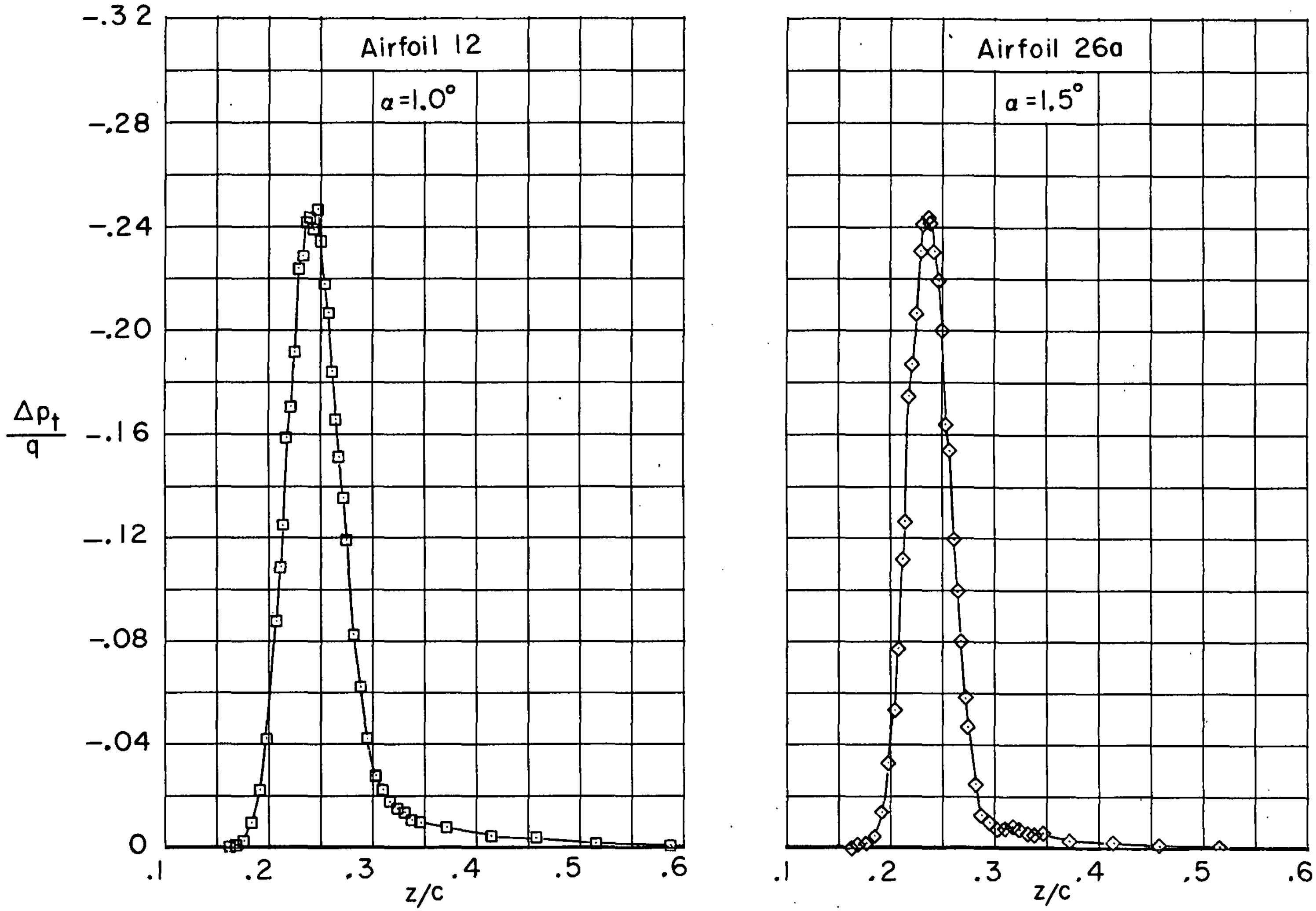
[redacted]



(c) $M = 0.78$; $c_n \approx 0.54$.

Figure 23.- Continued.

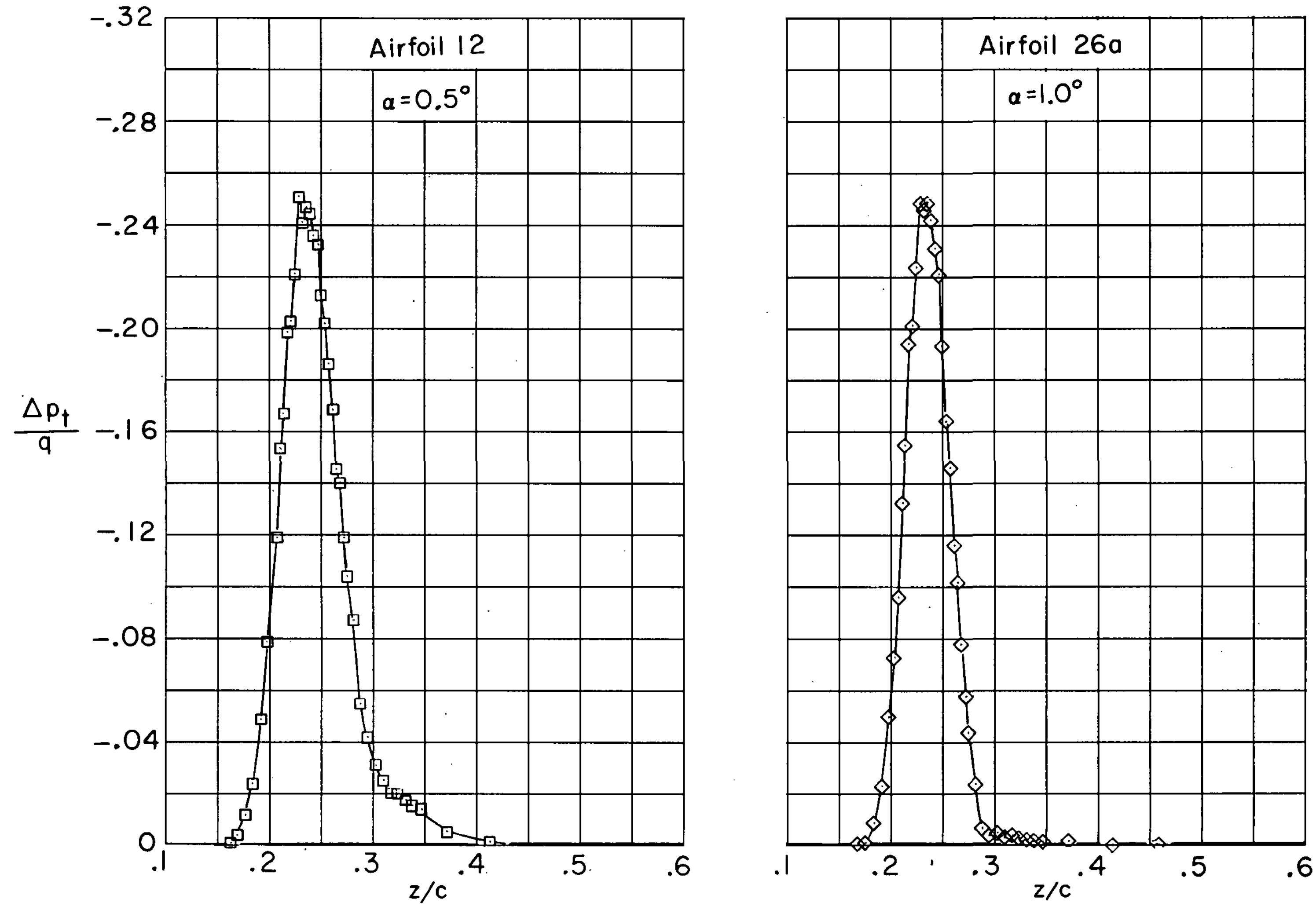
[redacted]



(d) $M = 0.79$; $c_n \approx 0.63$.

Figure 23.- Continued.

[redacted]

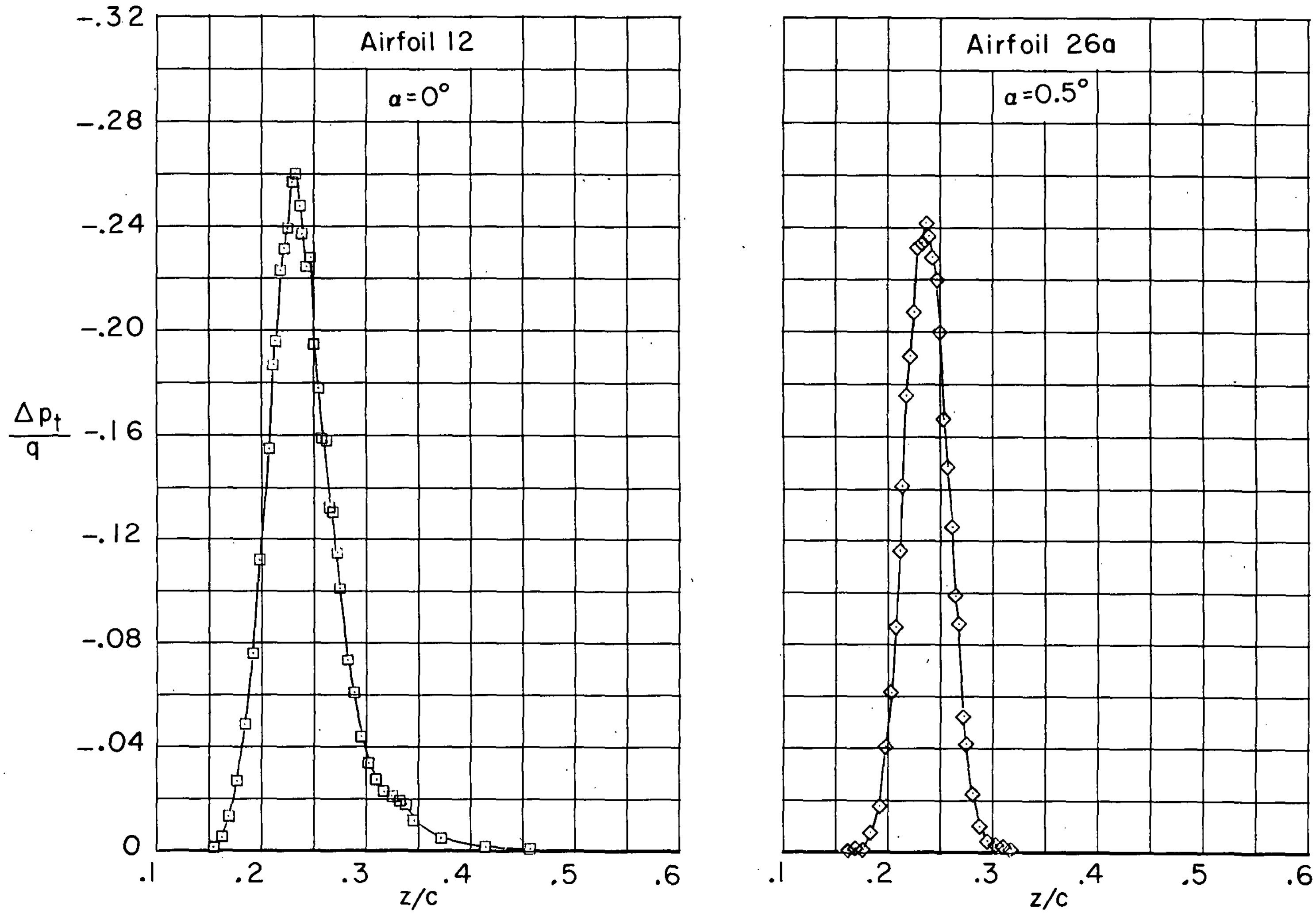


(e) $M = 0.80$; $c_n \approx 0.53$.

Figure 23.- Continued.

[redacted]

[redacted]

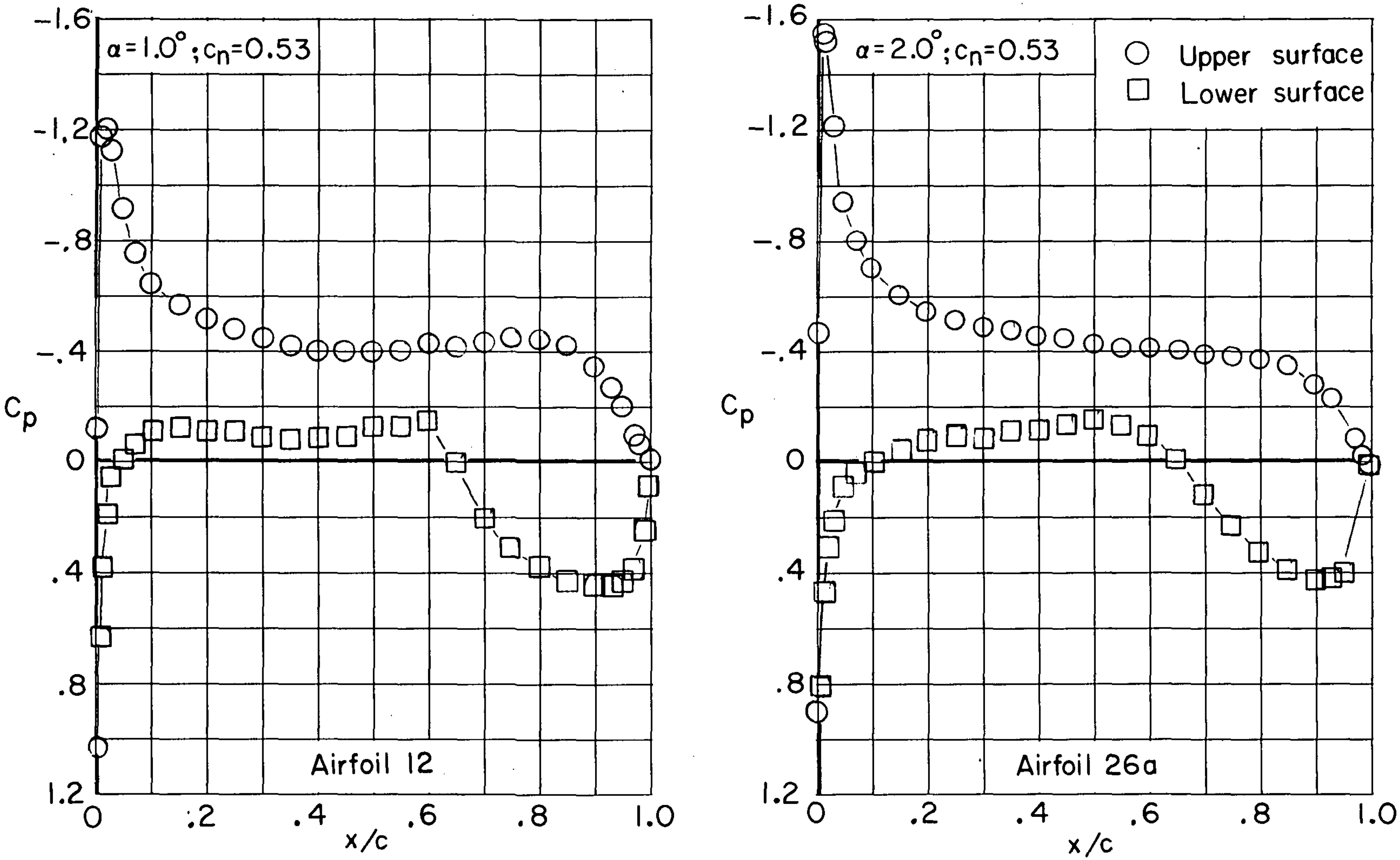


(f) $M = 0.81$; $c_n \approx 0.42$.

Figure 23.- Concluded.

[redacted]

[redacted]

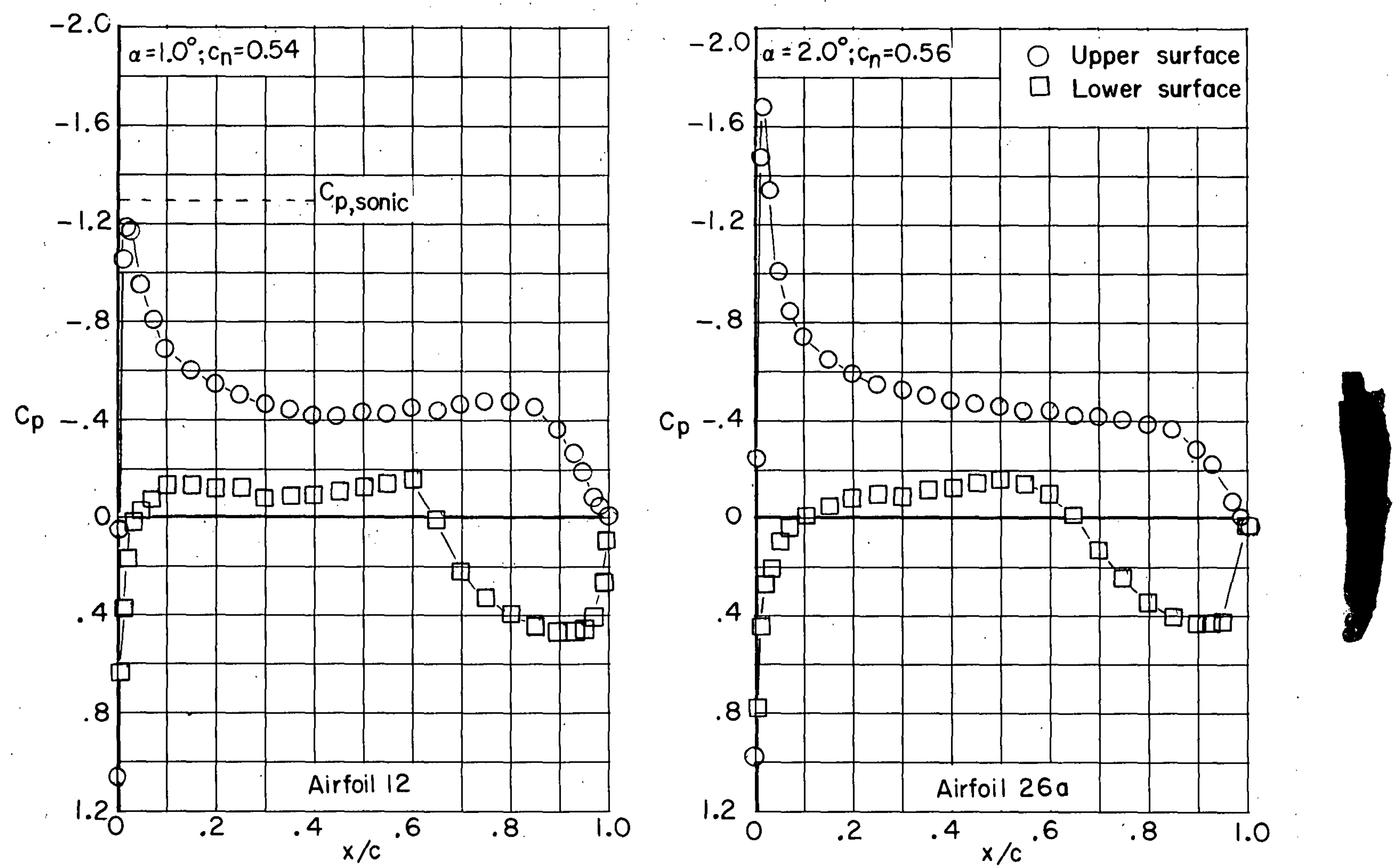


(a) M = 0.50.

Figure 24.- Chordwise pressure distribution for supercritical airfoils 12 and 26a at various Mach numbers and normal-force coefficients.

[redacted]

[redacted]

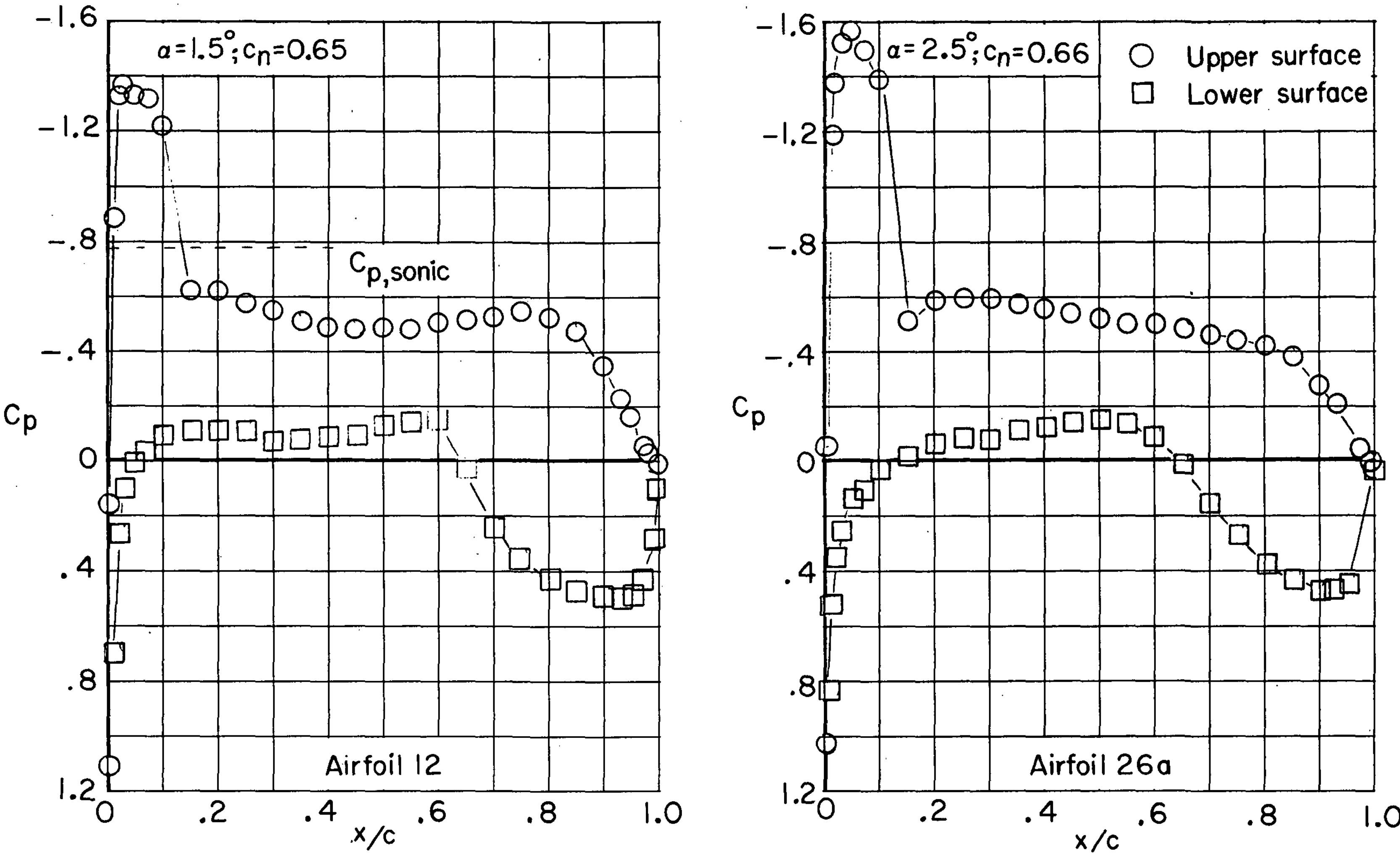


(b) M = 0.60.

Figure 24.- Continued.

[redacted]

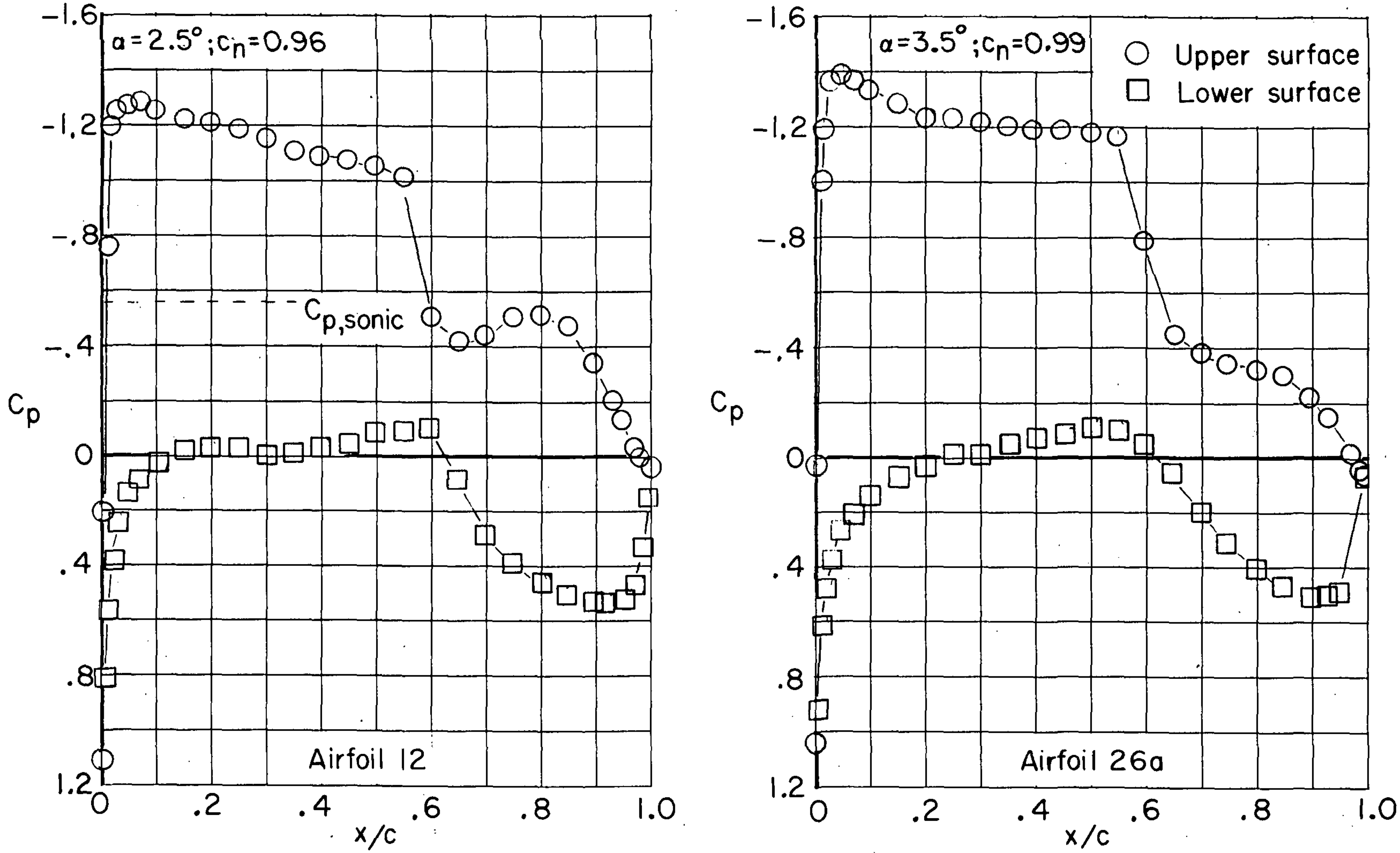
[redacted]



(c) M = 0.70.

Figure 24.- Continued.

[redacted]



(d) M = 0.76.

Figure 24.- Concluded.

FOR EARLY DOMESTIC DISSEMINATION

Because of its significant early commercial potential, this information, which has been developed under a U.S. Government program, is being disseminated within the United States in advance of general publication. This information may be duplicated and used by the recipient with the express limitation that it not be published. Release of this information to other domestic parties by the recipient shall be made subject to these limitations.

Foreign release may be made only with prior NASA approval and appropriate export licenses. This legend shall be marked on any reproduction of this information in whole or in part.

Review for general release September 30, 1982.

[redacted] CASE FILE
COPY

"The aeronautical and space activities of the United States shall be conducted so as to contribute . . . to the expansion of human knowledge of phenomena in the atmosphere and space. The Administration shall provide for the widest practicable and appropriate dissemination of information concerning its activities and the results thereof."

—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

NASA SCIENTIFIC AND TECHNICAL PUBLICATIONS

TECHNICAL REPORTS: Scientific and technical information considered important, complete, and a lasting contribution to existing knowledge.

TECHNICAL NOTES: Information less broad in scope but nevertheless of importance as a contribution to existing knowledge.

TECHNICAL MEMORANDUMS: Information receiving limited distribution because of preliminary data, security classification, or other reasons.

CONTRACTOR REPORTS: Scientific and technical information generated under a NASA contract or grant and considered an important contribution to existing knowledge.

TECHNICAL TRANSLATIONS: Information published in a foreign language considered to merit NASA distribution in English.

SPECIAL PUBLICATIONS: Information derived from or of value to NASA activities. Publications include conference proceedings, monographs, data compilations, handbooks, sourcebooks, and special bibliographies.

TECHNOLOGY UTILIZATION PUBLICATIONS: Information on technology used by NASA that may be of particular interest in commercial and other non-aerospace applications. Publications include Tech Briefs, Technology Utilization Reports, and Technology Surveys.

Details on the availability of these publications may be obtained from:

SCIENTIFIC AND TECHNICAL INFORMATION OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Washington, D.C. 20546

[redacted]