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THEORETICAL ANALYSIS OF AERODYNAMIC CHARACTERISTICS OF TWO HELICOPTER ROTOR AIRFOILS

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SUMMARY

An analytical study has been conducted to predict the aerodynamic characteristics of two helicopter rotor airfoils, versions of the FX69-HL-083 and FX71-H-080. Documentation of the predictive process covers the development of empirical factors used in conjunction with computer programs for airfoil analysis. Tables of lift, drag, and pitching-moment coefficient for each airfoil were prepared for two-dimensional, steady-flow conditions at Mach numbers from 0.3 to 0.9 and Reynolds numbers of 7.7 to 23.0×10^6, respectively.

INTRODUCTION

Tables of airfoil aerodynamic characteristics must be prepared in order to use rotorcraft computer programs to evaluate new rotor airfoils. These tables may be filled with results from wind-tunnel tests or analytical methods using computer programs for airfoils in two-dimensional, steady flow. Analytical methods can be used if the results of rotorcraft-performance programs are interpreted carefully, particularly when "low-confidence" portions of the airfoil tables have been utilized. Analytical methods generally offer an advantage in time, manpower, and costs when compared to wind-tunnel tests. However, wind-tunnel investigations still appear superior for the determination of parameters such as maximum lift coefficient at moderate Mach number.

This report contains the results of an analytical study to develop tables of the aerodynamic characteristics of modified FX69-HL-083 and FX71-H-080 airfoils. The subcritical and transonic airfoil programs of references 1 and 2, respectively, were used to provide basic coefficient values which were sometimes modified with empirical factors. As with previous studies, the task proceeded in three steps: first, evaluation of airfoil computer programs through correlation studies with existing experimental data; second, the development of empirical correction factors; and third, the application of the new total method. The only other documented case of such an analytical study for rotor airfoils is given in reference 3; this work on the NLR 7223-62 airfoil can be evaluated with information from references 4, 5, and 6.

SYMBOLS

c	airfoil chord, cm
c_d	section drag coefficient
c_ℓ	section lift coefficient
c_m	section pitching-moment coefficient, referenced to section quarter chord
C_p	pressure coefficient, $\frac{p - p_\infty}{q_\infty}$
$C_p{}^*$	pressure coefficient corresponding to local Mach number of 1.0
M	freestream Mach number
p	local static pressure at a point on airfoil, N/m^2

p_∞ freestream static pressure, N/m^2

q_∞ freestream dynamic pressure, N/m^2

R_c Reynolds number based on total chord and freestream conditions

t airfoil thickness, cm

x airfoil abscissa, cm

y airfoil ordinate, cm

y_c ordinate of airfoil mean line, cm

α angle of attack of airfoil reference line, deg

Subscripts

1,2 number of drag calculation method

ℓ lower surface

sep point of upper-surface separation

t point of transition to turbulent flow

u upper surface

AIRFOILS

Modified versions of the FX69-HL-083 and FX71-H-080, evaluated in this report, are identified as airfoils A and B, respectively. The modifications to each of the original shapes consist of changes to the leading-edge contour and the addition of trailing-edge reflex. Coordinates for airfoils A and B are listed in tables I and II; profiles are shown in figure 1. Thickness distributions and mean lines are shown in figure 2. For comparison, some geometric data are included for the NLR 7223-62 airfoil, also designated as the NLR-1. This airfoil was selected as the primary shape for correlation studies with computer-predicted characteristics and experimental data.

Calculated pressure distributions for these three airfoils are presented for selected conditions in figure 3. These conditions are representative of combinations of Mach number and lift coefficient that are reached by airfoil sections near to the outboard end of a rotor blade. Similarities in the pressure distributions for the NLR-1 and B airfoils can be attributed to the commonality of design emphasis: both were developed to achieve high values of drag-divergence Mach number at low lift coefficients. The pressure distribution of airfoil A in figure 3(a) has a large region of favorable pressure gradient that reflects a design emphasis on laminar flow. Another major difference between airfoils A and B is the difference in leading-edge suction peaks reached at high lift coefficients. This difference is clearly seen in figure 3(b); in figure 3(c), the lowest C_p value on airfoil B is 1.0 lower than the lowest on airfoil A. Thus, at the same lift coefficient and Mach number, airfoil B will have higher local induced velocities, which make that airfoil more susceptible to shock stall.

ANALYTICAL METHODS

Computer Programs

Two computer programs for airfoil analysis were used. One was the subcritical, viscous-flow program developed at North Carolina State University and described in reference 1; this program will be referred to as the NCSU program. It is an improved, single-element version of the program described in reference 7. The second major computational tool was the viscous, transonic-flow program developed at New York University; this program, which is described in reference 2, is referred to as the NYU program. Both programs calculate pressure distribution and coefficients of lift, drag, and pitching moment;

both require iterations between potential flow solutions for a "fluid" airfoil and boundary-layer calculations for each test condition.

NCSU Program

The NCSU program is generally appropriate for utilization in cases where the local flow field remains subcritical. Although this program cannot calculate through separated-flow regions (bubbles or trailing-edge separation), it indicates that various types of separation may occur and then completes calculations for attached flow. This program was run at appropriate Reynolds numbers with free transition. The resulting calculations of laminar-to-turbulent transition point influenced the selection of the transition-point inputs for the NYU program.

NYU Program

The NYU program was utilized at conditions with Mach numbers equal to or greater than 0.4. This program is very useful for predicting airfoil characteristics either at the low-lift, high Mach number conditions or near to the stall boundaries for moderate Mach numbers. Upper- and lower-surface transition points specified by the user are the starting points for the purely turbulent boundary layer. This program also gives indications of conditions that could induce separation but it calculates as if no separated flow existed.

At least two features of the NYU program are significant for use with rotorcraft airfoils. First, a program option must be used to allow utilization of the same boundary layer equations for upper and lower surfaces. (This is in contrast to the needs of a correlation study, such as that of reference 8, which is concerned with a supercritical airfoil.) The second feature is the calculation scheme that "marches" around the airfoil in one continuous path:

trailing to leading edge on the lower surface, followed by leading to trailing edge on the upper surface. For cases with negative lift at high Mach numbers, the NYU program sometimes required that an inverted set of coordinates were input: the equations then moved through the strong shocks, located on the true lower surface, in the numerically desirable direction.

Correlation Studies

Correlation with experimental data and the application of resulting empiricism are required when using the NCSU and NYU programs at the conditions required for rotor airfoils. Initial studies were completed with data for a wide variety of such airfoils (refs. 5, 6, 9, 10, and 11). The final work relied more on results from well-documented tests of the NLR-1 airfoil. Only a small sample of the correlation results are presented in this report. Also, some appropriate comparisons between theory and experimental data are contained in references 1 and 2.

Figures 4 and 5 present a typical comparison of computer-program predictions and wind-tunnel results for Mach numbers of 0.4 and 0.5. In figure 4, pitching-moment coefficient predicted for the NLR-1 is too positive in comparison to wind-tunnel results; also, the lift-curve slope appears to be slightly high. In figure 5, results from the NYU program are compared to data from reference 10. (Data from reference 5 are also included even though taken at a higher Reynolds number.) The trends noted in the preceding figure are seen here: at constant Reynolds number, theory predicts pitching-moment coefficients that are too positive and lift-coefficients that are too large. The drag-coefficient comparison is typical of other comparisons not presented here. Rotor-airfoil data generally showed better correlation with the "old" drag values (denoted in the figure as c_{d_1}) of the NYU program.

Figure 6 presents the results of both NYU program calculations and wind-tunnel tests of the NLR-1 airfoil at Mach numbers of 0.6 and 0.7. The correlation between theory and experiment is better for lift and pitching-moment coefficient at these higher Mach numbers. However, the trend for drag-coefficient correlation is the same as in figure 5: c_{d_1}, "old" drag computed with a "crude" (80 by 15 point) calculation grid, produced better correlation with experimental data.

Predictive Analysis

A set of analytical methods were established as a result of the total correlation study. The NCSU program was utilized, with free transition, for Mach numbers of 0.3 and 0.4. For these conditions, the value of pitching-moment coefficient in the tables is a maximum of 0.01 greater than the calculated value. The NYU program was used for Mach numbers greater than 0.4. Below a Mach number of 0.7, program-computed results were adjusted by factors derived through correlation with the NLR-1 airfoil: the angle of zero lift remained unchanged, lift-curve slope was decreased slightly, and pitching-moment coefficient was decremented by a maximum of 0.02. Drag coefficients in the table from the NYU program are the "old" drag values computed with a "crude" grid. Since no reliable method of stall prediction was developed, the lift coefficient data estimated for angles of attack beyond indications of stall are labeled as "low-confidence" data.

The analytical methods described above were applied to airfoils A and B. The selection of a 1.07 m chord and nominal sea-level atmospheric conditions resulted in a ratio of Reynolds number to Mach number of 25.5×10^6. This gives a Reynolds number of 17.1×10^6 at a representative hover tip Mach number of 0.67.

RESULTS

The results of this study are the prediction of airfoil characteristics for airfoils A and B in steady, two-dimensional flow. These results are listed in tables III and IV for airfoils A and B, respectively. The same values are presented in figures 7, 8, and 9. The boundary for low-confidence lift data is indicated in figure 7. No estimates were prepared on the effects of the three-dimensional, unsteady-flow environment of the rotor.

The tables were formatted to facilitate their use in rotor-performance programs. Most rotorcraft programs require airfoil tables to extend from Mach 0.0 to 1.0 and angles of attack from -180 degrees to +180 degrees. It is suggested that the remainder of the airfoil tables for the programs can be filled with NACA 0012 data adjusted as indicated in reference 9.

CONCLUDING REMARKS

Tables of aerodynamic coefficients for two rotorcraft airfoils, versions of the FX69-HL-083 and FX71-H-080, were prepared with analytical methods. Correlation work indicated that correction factors should be applied to the results of calculations of aerodynamic characteristics by airfoil-analysis computer programs. These factors were developed and applied to programs for airfoil analysis to obtain the predicted airfoil characteristics.

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TABLE I.- DESIGN COORDINATES FOR AIRFOIL A

x/c	y_u/c	y_ℓ/c
0.000000	0.000000	0.000000
.001070	.002970	-.003910
.004280	.006850	-.006160
.009610	.011150	-.007380
.017040	.015740	-.008650
.026540	.020440	-.009790
.038060	.025240	-.011100
.051560	.030010	-.012300
.066990	.034700	-.013400
.084270	.039260	-.014380
.103320	.043670	-.015180
.124080	.047790	-.015810
.146450	.051600	-.016340
.170330	.055000	-.016730
.195620	.057960	-.017100
.222220	.060410	-.017390
.250000	.062390	-.017700
.278860	.063810	-.017960
.308660	.064680	-.018260
.339280	.064970	-.018520
.370590	.064760	-.018800
.402460	.064030	-.019040
.434740	.062870	-.019240
.467300	.061220	-.019350
.500000	.059160	-.019390
.532700	.056620	-.019250
.565260	.053700	-.019020
.597550	.050400	-.018700
.629410	.046840	-.018330
.660720	.043050	-.017890
.691340	.039110	-.017400
.721140	.035050	-.016840
.750000	.030960	-.016240
.777780	.026890	-.015580
.804380	.022920	-.014870
.829670	.019090	-.014080
.853550	.015510	-.013190
.875920	.011910	-.012200
.915730	.006060	-.009860
.948440	.002780	-.007070
.973460	.001230	-.004190
.990390	.000720	-.002000
.998930	.000530	-.000680
1.000000	0.000000	0.000000

Leading-edge radius: 0.00578 c

TABLE II.- DESIGN COORDINATES FOR AIRFOIL B

x/c	y_u/c	y_ℓ/c
0.000000	0.000000	0.000000
.001070	.004630	-.003060
.004280	.008960	-.005790
.009610	.013240	-.008280
.017040	.017700	-.010630
.026540	.022270	-.012840
.038060	.026690	-.014860
.051560	.030760	-.016700
.066990	.034520	-.018430
.084270	.037870	-.020030
.103320	.040710	-.021450
.124080	.043240	-.022770
.146450	.045420	-.023940
.170330	.047240	-.024930
.195620	.048720	-.025740
.222220	.049930	-.026390
.250000	.050860	-.026930
.278860	.051510	-.027380
.308660	.051860	-.027770
.339280	.051940	-.028040
.370590	.051840	-.028110
.402460	.051430	-.027960
.434740	.050670	-.027670
.467300	.049570	-.027260
.500000	.048140	-.026740
.532700	.046380	-.026130
.565260	.044310	-.025400
.597550	.041950	-.024550
.629410	.039320	-.023570
.660720	.036460	-.022480
.691340	.033400	-.021280
.721140	.030200	-.020000
.750000	.026920	-.018670
.777780	.023610	-.017310
.804380	.020330	-.015950
.829670	.017150	-.014620
.853550	.014070	-.013330
.875920	.011200	-.012090
.915730	.006700	-.009790
.948440	.003630	-.007450
.973460	.002020	-.004870
.990390	.001360	-.002530
.998930	.000930	-.000620
1.000000	0.000000	0.000000

Leading-edge radius: 0.00718 c

TABLE III.- AIRFOIL A CHARACTERISTICS

M	α	c_ℓ	α	c_d	α	c_m
0.3	0.0	0.11	0.0	0.0055	0.0	-0.013
	10.0	1.08	3.0	0.0048	5.0	-0.017
	11.0	1.10	5.0	0.0058	9.0	-0.019
			10.0	0.0105	10.5	-0.019
			11.0	0.120	11.5	-0.016
					13.0	-0.023
0.4	0.0	0.11	0.0	0.0053	0.0	-0.013
	8.0	0.96	3.0	0.0050	5.0	-0.017
	9.0	1.05	5.0	0.0058	8.5	-0.018
	10.0	1.10	8.0	0.0090	10.0	-0.016
			9.0	0.0130	11.0	-0.012
					12.0	-0.022
0.5	0.0	0.12	0.0	0.006	0.0	-0.014
	8.0	1.05	5.0	0.006	6.0	-0.022
	9.0	1.1	7.0	0.009	7.0	-0.020
			8.0	0.012	8.0	-0.018
0.6	0.0	0.13	0.0	0.006	0.0	-0.014
	7.0	1.00	5.0	0.006	5.0	-0.015
	9.0	1.10	6.0	0.009	6.0	-0.01
			7.0	0.018	7.0	-0.01
			8.0	0.030		
0.65	0.0	0.14	0.0	0.006	0.0	-0.015
	6.0	0.99	2.0	0.007	3.0	-0.016
	8.0	1.10	4.0	0.009	5.0	-0.012
			5.0	0.017	7.0	-0.017
			6.0	0.031	8.0	-0.026
			7.0	0.050		
			8.0	0.080		
0.7	0.0	0.14	0.0	0.006	0.0	-0.015
	5.0	0.91	2.0	0.008	5.0	-0.015
	6.0	1.05	3.0	0.012	6.0	-0.022
	7.0	1.10	4.0	0.024		
			5.0	0.043		
			7.0	0.090		

TABLE III.- AIRFOIL A CHARACTERISTICS (CONCLUDED)

M	α	c_ℓ	α	c_d	α	c_m
0.8	-2.7 -1.0 0.0 1.0 2.0	-0.30 0.03 0.24 0.43 0.60	-2.7 -2.0 -1.0 0.0 1.0 2.0 3.0	0.023 0.015 0.011 0.016 0.027 0.043 0.065	-2.7 -1.0 0.0 1.0 2.0	-0.019 -0.022 -0.035 -0.060 -0.080
0.825			-2.6 -2.0 -1.0 0.0 1.0 2.0 3.0	0.026 0.019 0.016 0.023 0.036 0.054 0.076	-2.6 -2.0 0.0 2.0	-0.017 -0.022 -0.051 -0.101
0.85	-2.4 0.0 1.0 2.0	-0.30 0.21 0.40 0.55	-2.4 -2.0 -1.0 0.0 1.0 2.0 3.0	0.032 0.027 0.024 0.030 0.045 0.062 0.083	-2.4 -2.0 0.0 2.0	-0.022 -0.04 -0.067 -0.120
0.875	-2.0 0.0 1.0	-0.23 0.18 0.34	-2.0 -1.0 0.0 1.0 2.0 3.0	0.037 0.033 0.037 0.051 0.068 0.090	-2.0 -1.0 0.0 1.0	-0.027 -0.064 -0.084 -0.100
0.9	-2.0 -1.0 0.0 1.0	-0.30 -0.05 0.12 0.27	-2.0 -1.0 0.0 1.0 2.0 3.0	0.047 0.042 0.044 0.057 0.074 0.097	-2.0 0.0 1.0	0.005 -0.100 -0.105

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TABLE IV.- AIRFOIL B CHARACTERISTICS

M	α	c_ℓ	α	c_d	α	c_m
0.3	0.0	0.04	0.0	0.0058	0.0	-0.003
	11.0	1.15	3.0	0.0052	9.0	-0.007
	12.5	1.20	5.0	0.0059	10.5	-0.006
			9.0	0.0105	11.5	-0.003
			11.0	0.0135	12.0	-0.003
			12.5	0.0165	13.0	-0.008
0.4	0.0	0.04	0.0	0.0057	0.0	-0.003
	9.0	1.00	3.0	0.0052	5.0	-0.005
	10.5	1.08	5.0	0.006	8.5	-0.005
			7.5	0.009	10.0	-0.001
			8.5	0.013	10.5	0.002
					11.0	0.0
0.5	0.0	0.05	0.0	0.006	0.0	-0.003
	7.0	0.85	5.0	0.006	6.0	-0.007
	8.0	0.95	7.0	0.010	7.0	-0.003
	9.0	1.00	8.0	0.014	8.0	-0.001
0.6	0.0	0.06	0.0	0.006	0.0	-0.003
	7.0	0.90	4.0	0.006	5.0	0.002
	8.0	0.97	5.0	0.008	6.0	0.005
	9.0	1.00	6.0	0.015	7.0	0.007
			8.0	0.047	8.0	0.007
0.65	0.0	0.06	0.0	0.006	0.0	-0.003
	7.0	1.01	2.0	0.007	2.0	-0.002
	9.0	1.11	4.0	0.012	6.0	0.009
			5.0	0.023	8.0	0.001
			6.0	0.038		
			7.0	0.060		
			8.0	0.090		
0.7	0.0	0.08	0.0	0.006	0.0	-0.003
	5.0	0.84	2.0	0.007	3.0	-0.003
	7.0	1.00	3.0	0.010	5.0	0.004
			4.0	0.020	7.0	-0.015
			5.0	0.040		
			7.0	0.090		

TABLE IV.- AIRFOIL B CHARACTERISTICS (CONCLUDED)

M	α	c_ℓ	α	c_d	α	c_m
0.8	-2.0	-0.27	-2.0	0.012	-2.0	-0.013
	-1.0	-0.07	-1.0	0.007	-1.0	-0.009
	0.0	0.13	0.0	0.007	0.0	-0.001
	1.0	0.32	1.0	0.014	1.0	-0.007
	2.0	0.48	2.0	0.025	2.0	-0.018
			3.0	0.045		
0.825			-2.0	0.019	-2.0	-0.005
			-1.0	0.011	-1.0	0.0
			0.0	0.015	0.0	-0.012
			1.0	0.020	1.0	-0.024
			2.0	0.035	2.0	-0.046
			3.0	0.058		
0.85	-2.0	-0.30	-2.0	0.027	-2.0	0.01
	0.0	0.12	-1.0	0.019	-1.0	-0.009
	2.0	0.51	0.0	0.021	0.0	-0.022
			1.0	0.027	1.0	-0.041
			2.0	0.045	2.0	-0.075
			3.0	0.070		
0.875	-2.0	-0.30	-2.0	0.027	-2.0	0.024
	0.0	0.09	-1.0	0.019	-1.0	-0[illegible]12
	1.0	0.32	0.0	0.021	0.0	-0.033
			1.0	0.033	1.0	-0.070
			2.0	0.052		
			3.0	0.073		
0.9	-2.0	-0.30	-2.0	0.039	-2.0	0.037
	-1.0	-0.13	-1.0	0.031	0.0	-0.043
	0.0	0.06	0.0	0.031	1.0	-0.098
			1.0	0.043		
			2.0	0.059		
			3.0	0.077		

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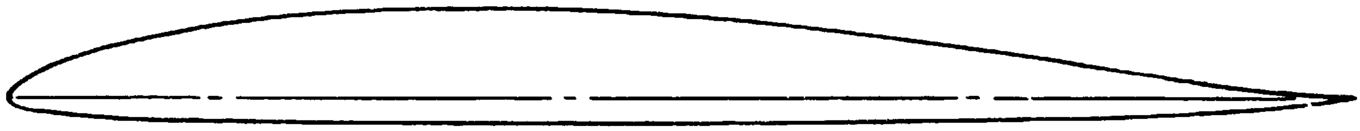
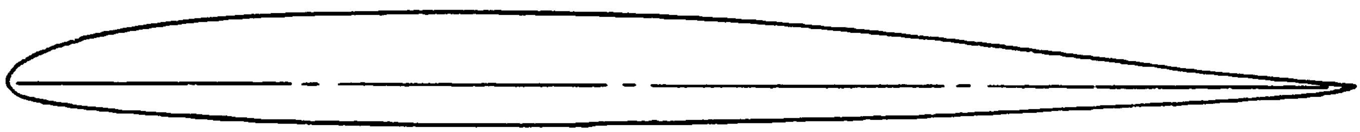
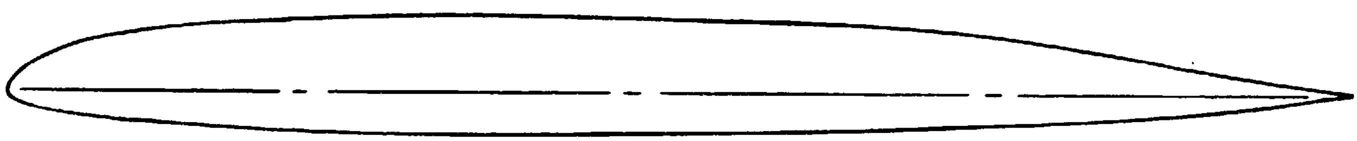


Airfoil A (FX69-HL-083 modified)



Airfoil B (FX71-H-080 modified)



NLR-1

Figure 1.- Airfoil profiles.

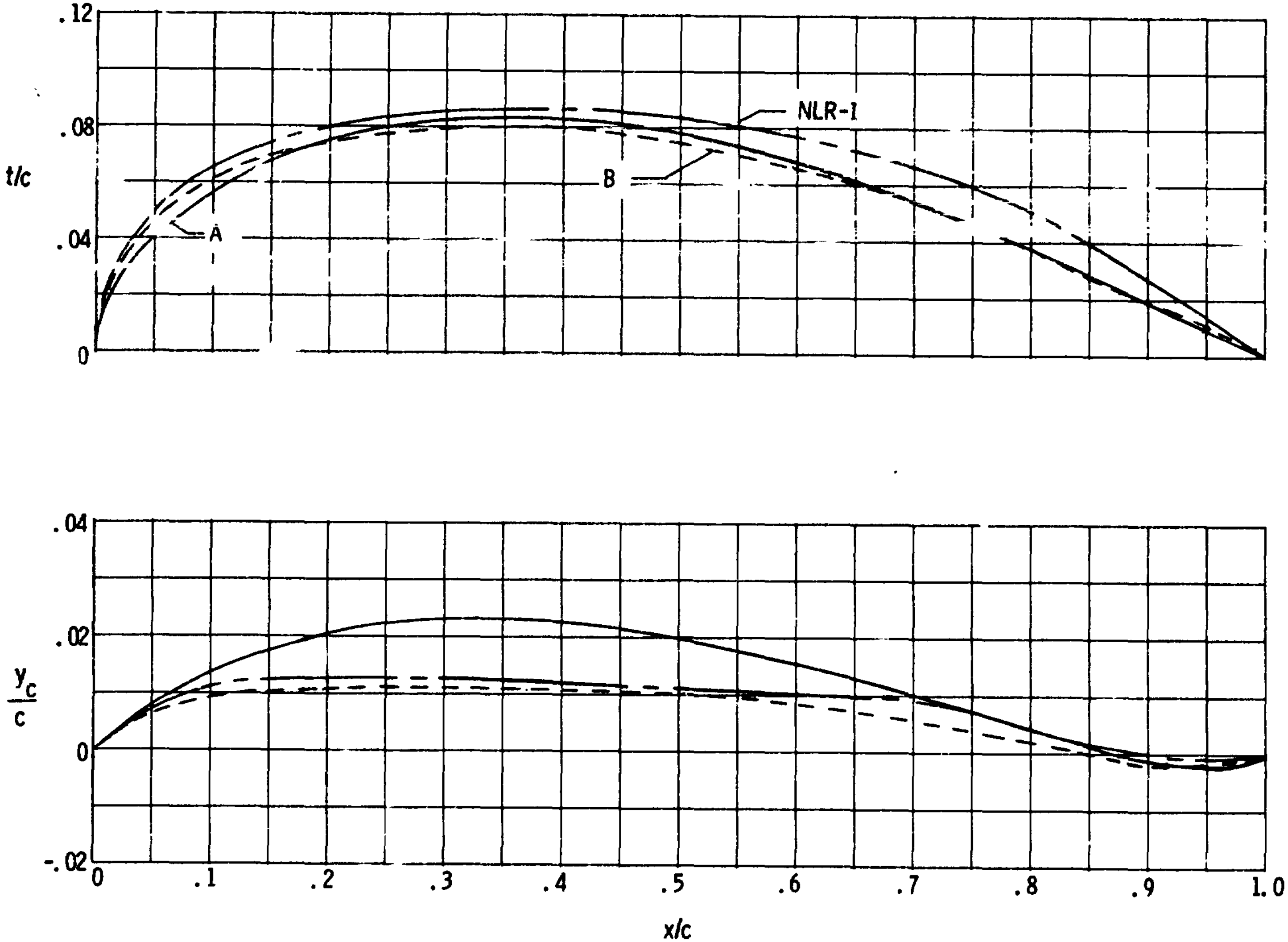
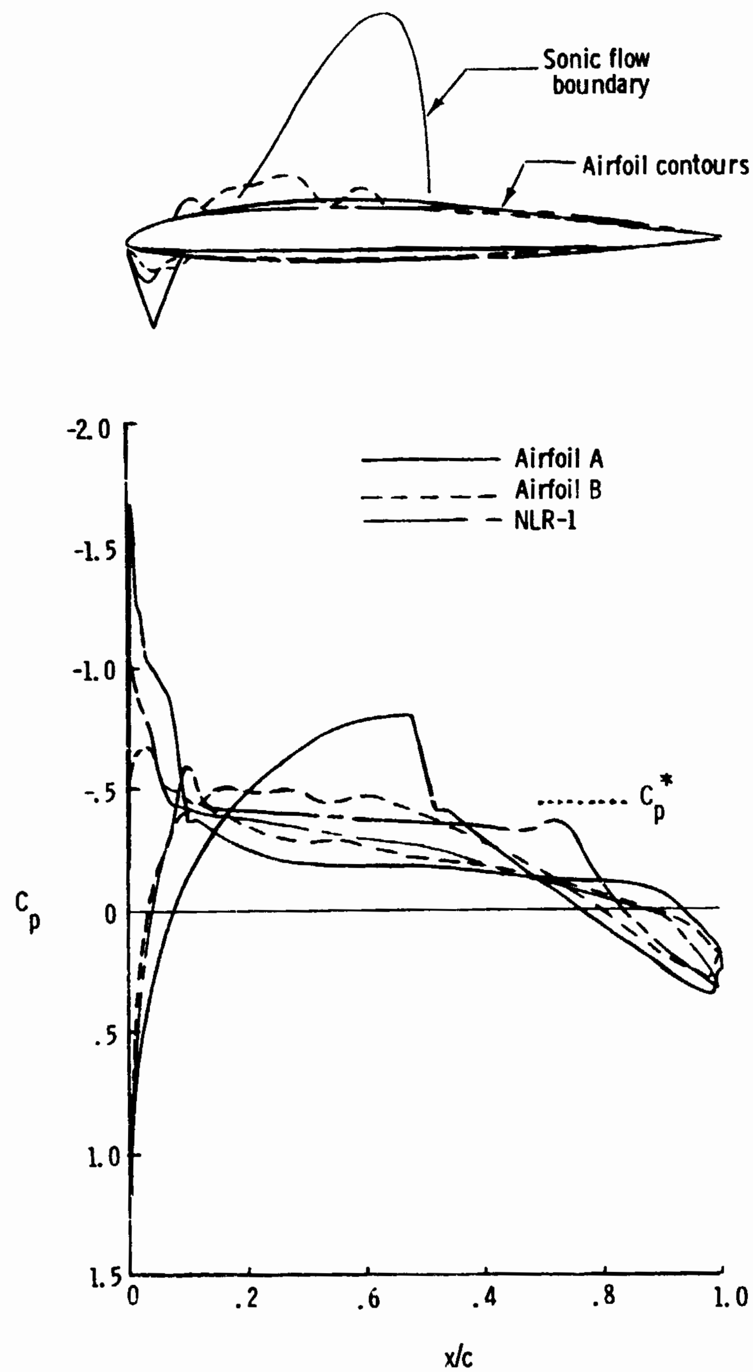
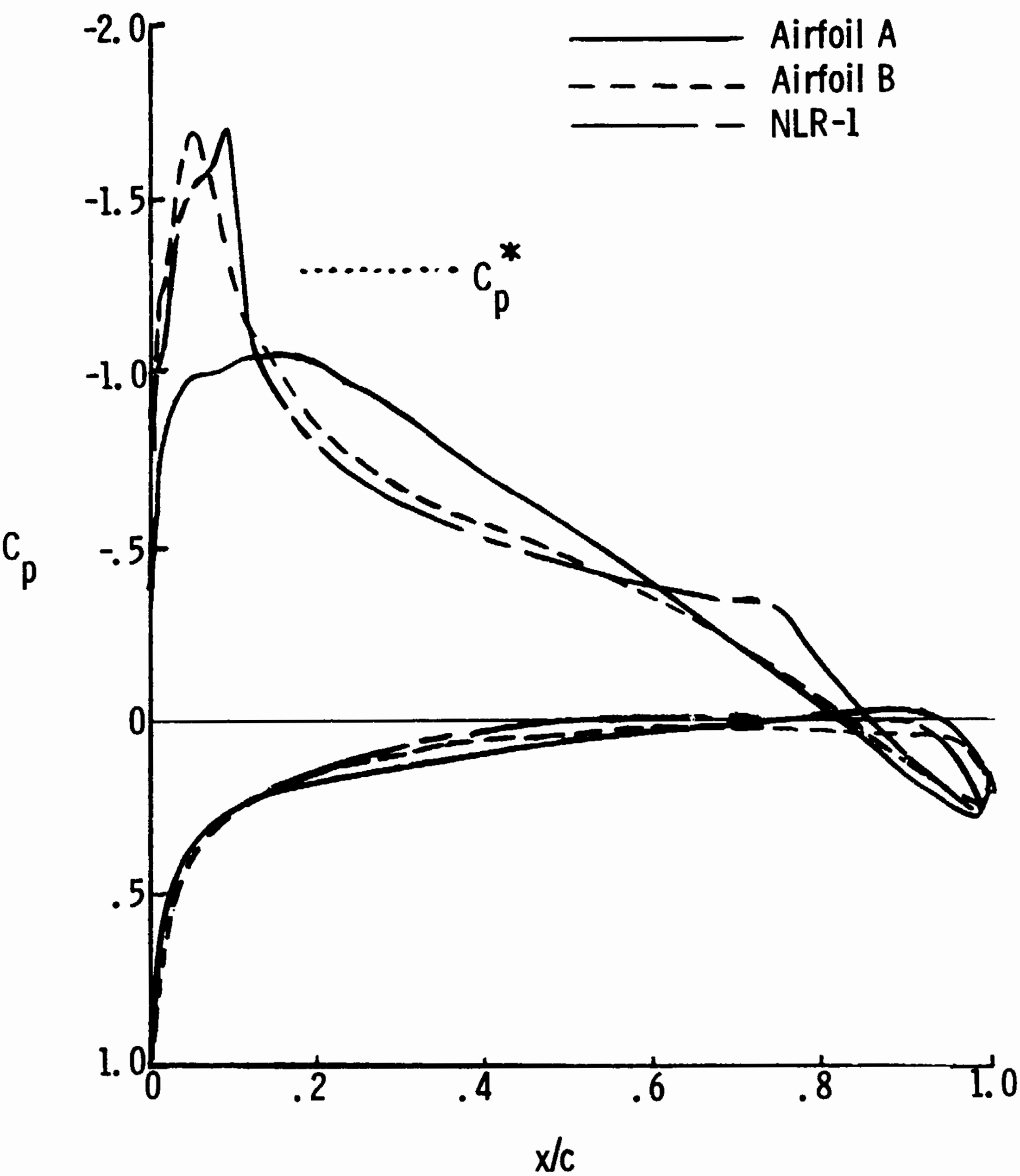


Figure 2.- Airfoil thickness distributions and mean lines.



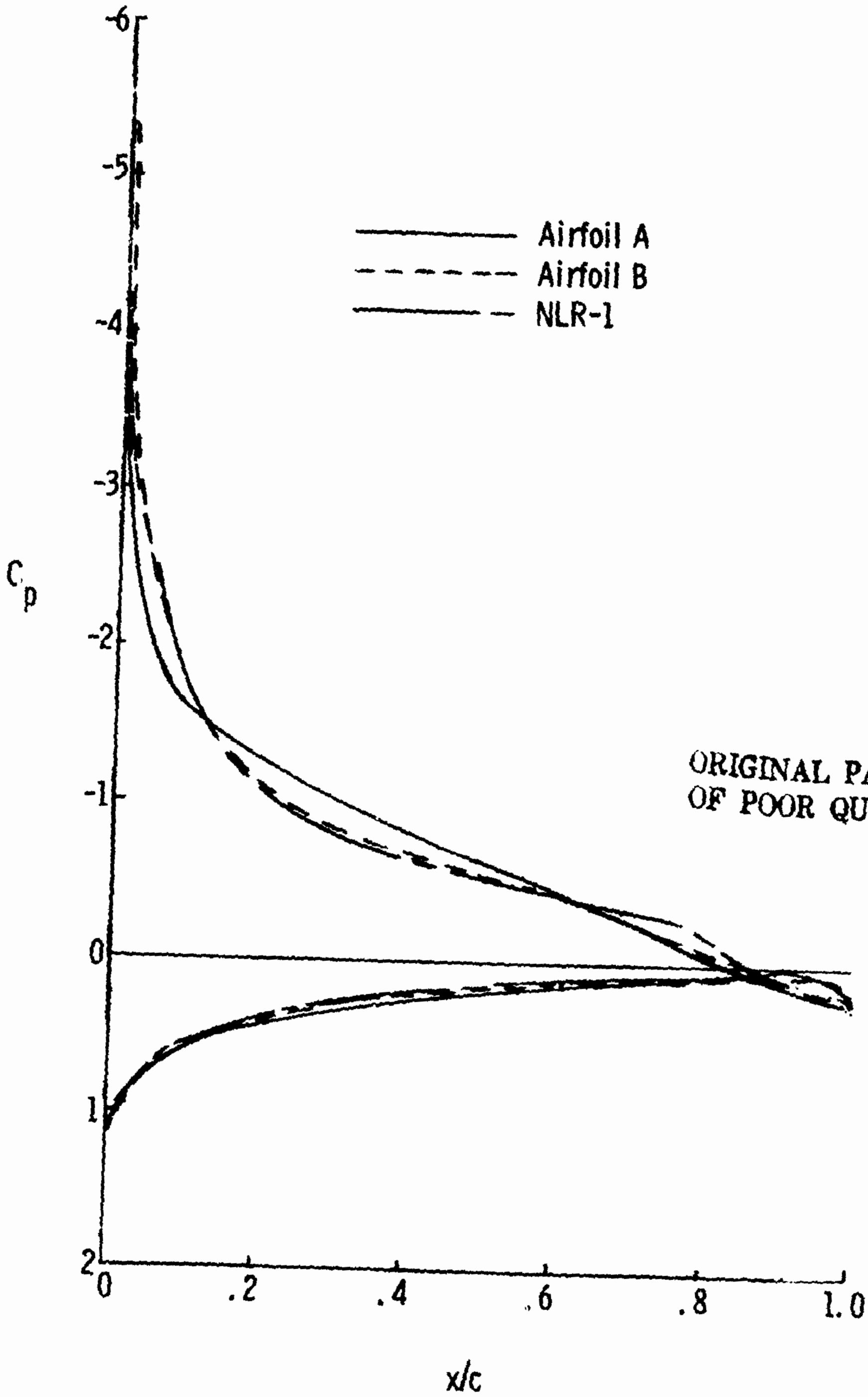
(a) $M = 0.8$; $c_\ell = 0.0$; $R_c = 20.4 \times 10^6$.

Figure 3.- Pressure distributions and sonic-flow boundaries of A, B, and NLR-1 airfoils calculated with NYU program.



(b) M = 0.6; c_ℓ = 0.6; R_c = 15.3 x 10^6.

Figure 3.- Continued.



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(c) $M = 0.4$; $c_{\ell} = 1.0$; $R_c = 10.2 \times 10^6$.

Figure 3.- Concluded.

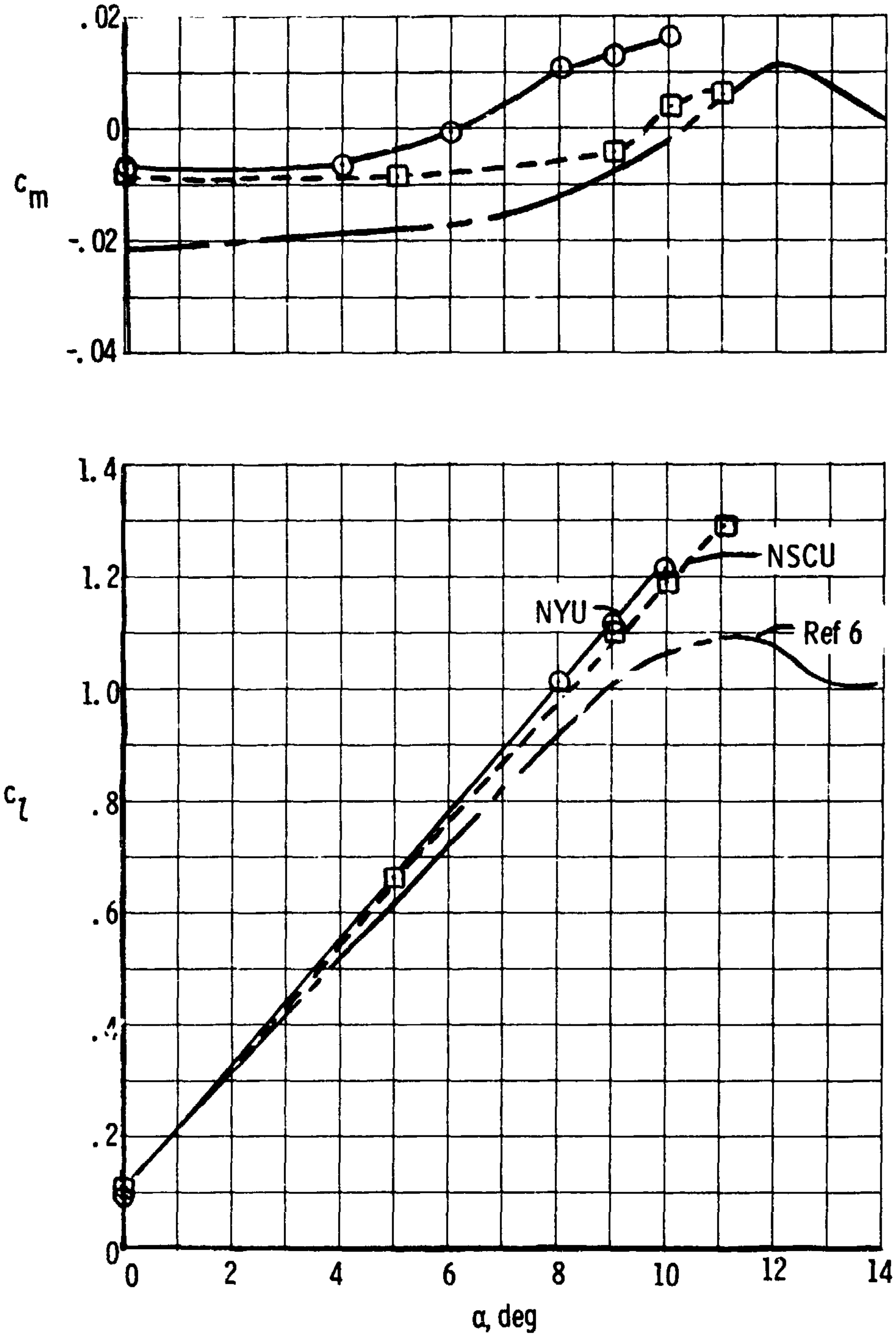
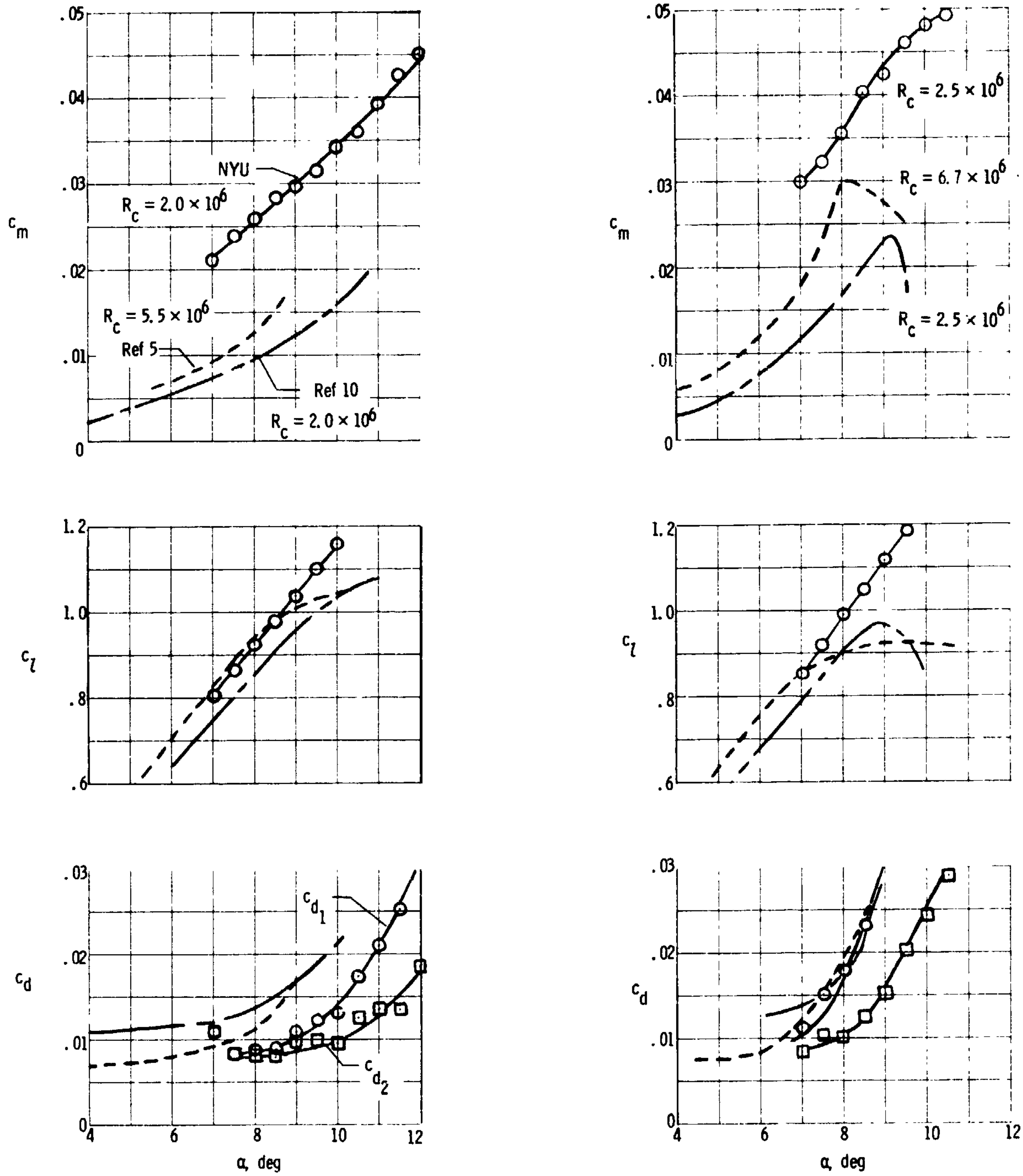


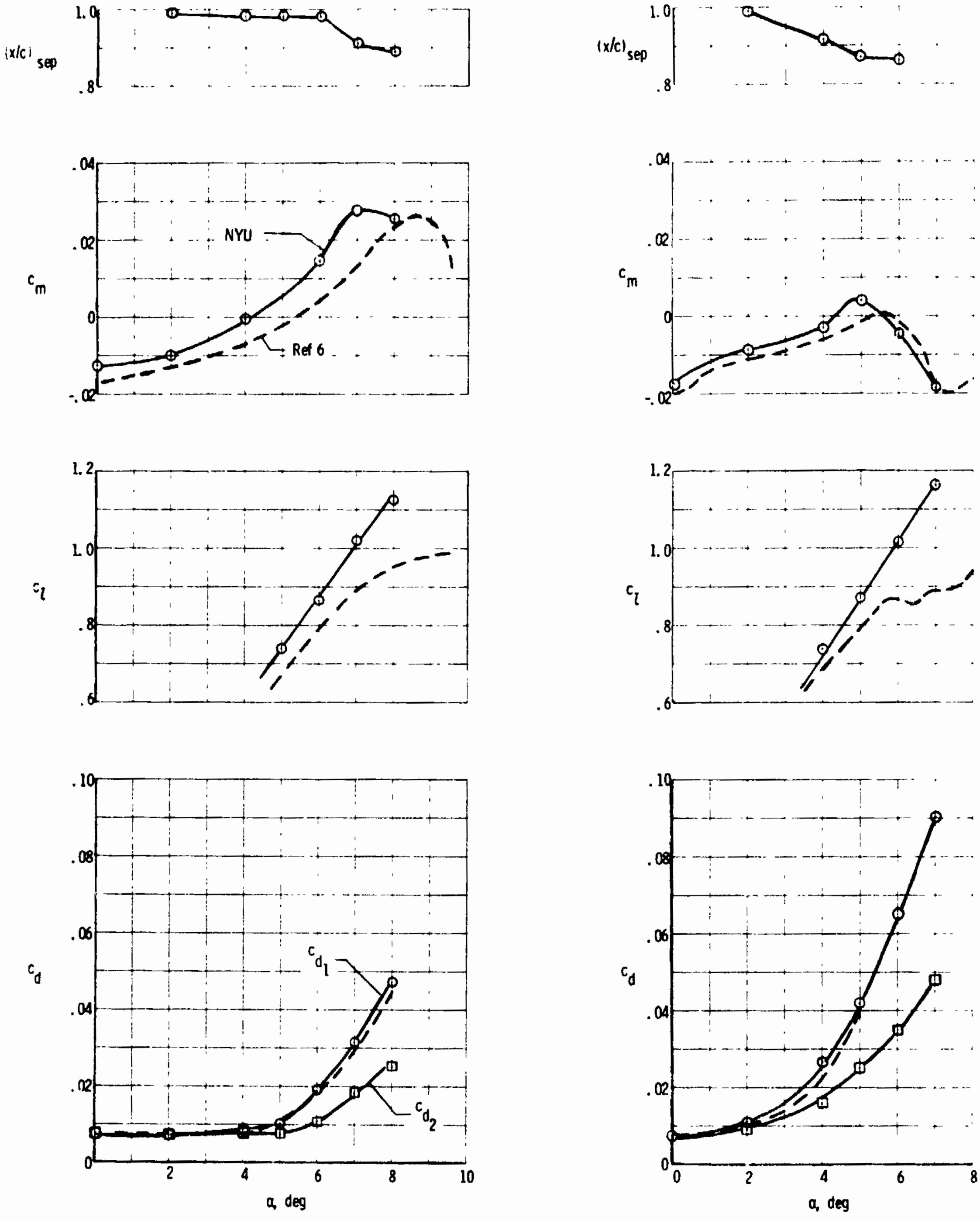
Figure 4.- Calculated and experimental values of aerodynamic characteristics of NLR-1 airfoil at $M = 0.4$, $R_c = 6.5 \times 10^6$.



(a) M = 0.4.

(b) M = 0.5.

Figure 5.- Calculated and experimental values of aerodynamic characteristics of NACA 0012 airfoil.



(a) M = 0.6
R_c = 9.6 x 10^6.

(b) M = 0.7
R_c = 10.4 x 10^6.

Figure 6.- Calculated and experimental values of aerodynamic characteristics of NLR-1 airfoil at high speed.

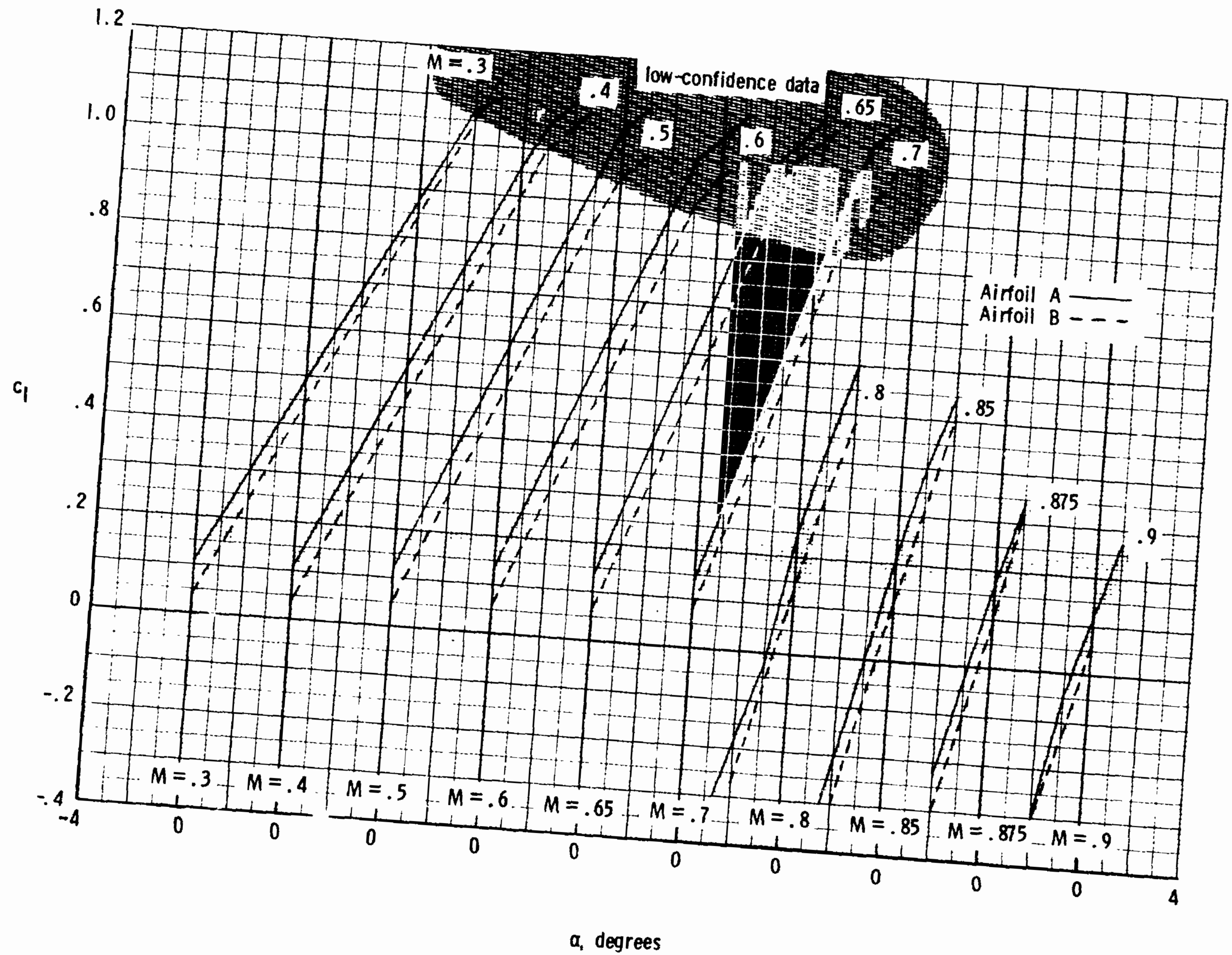


Figure 7.- Predicted lift coefficient.

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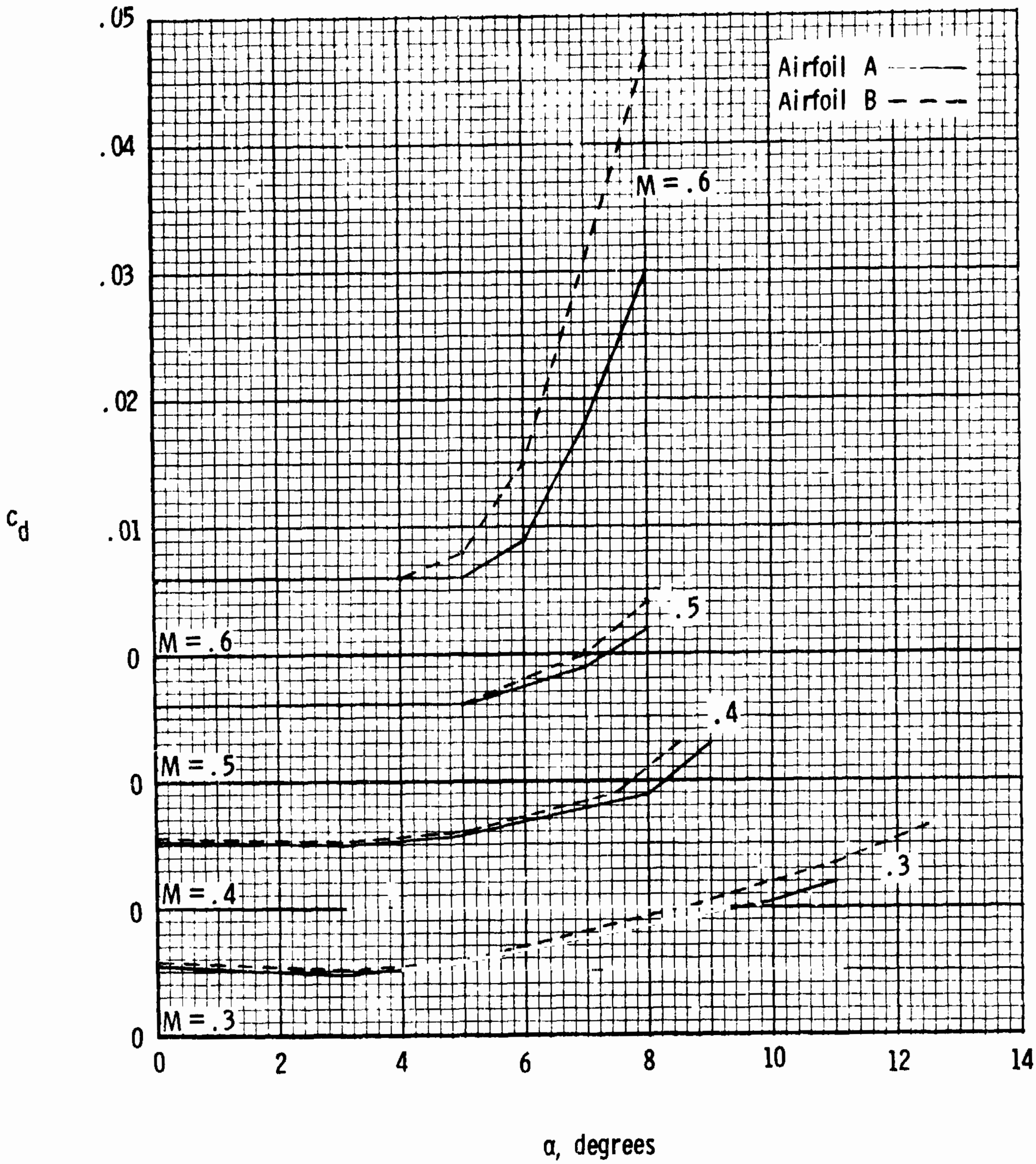

Figure 8.- Predicted drag coefficient.

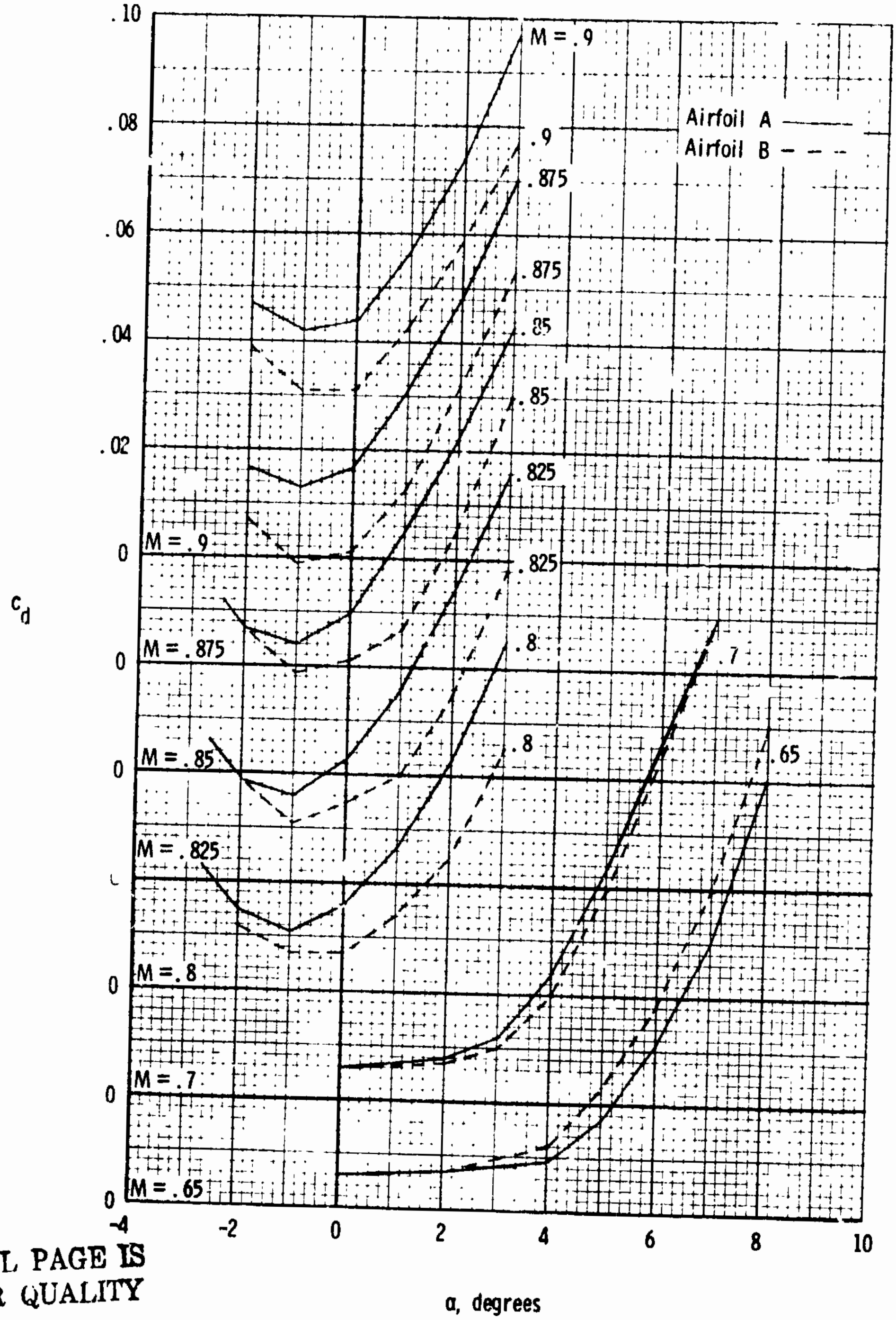

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Figure 8.- Concluded.

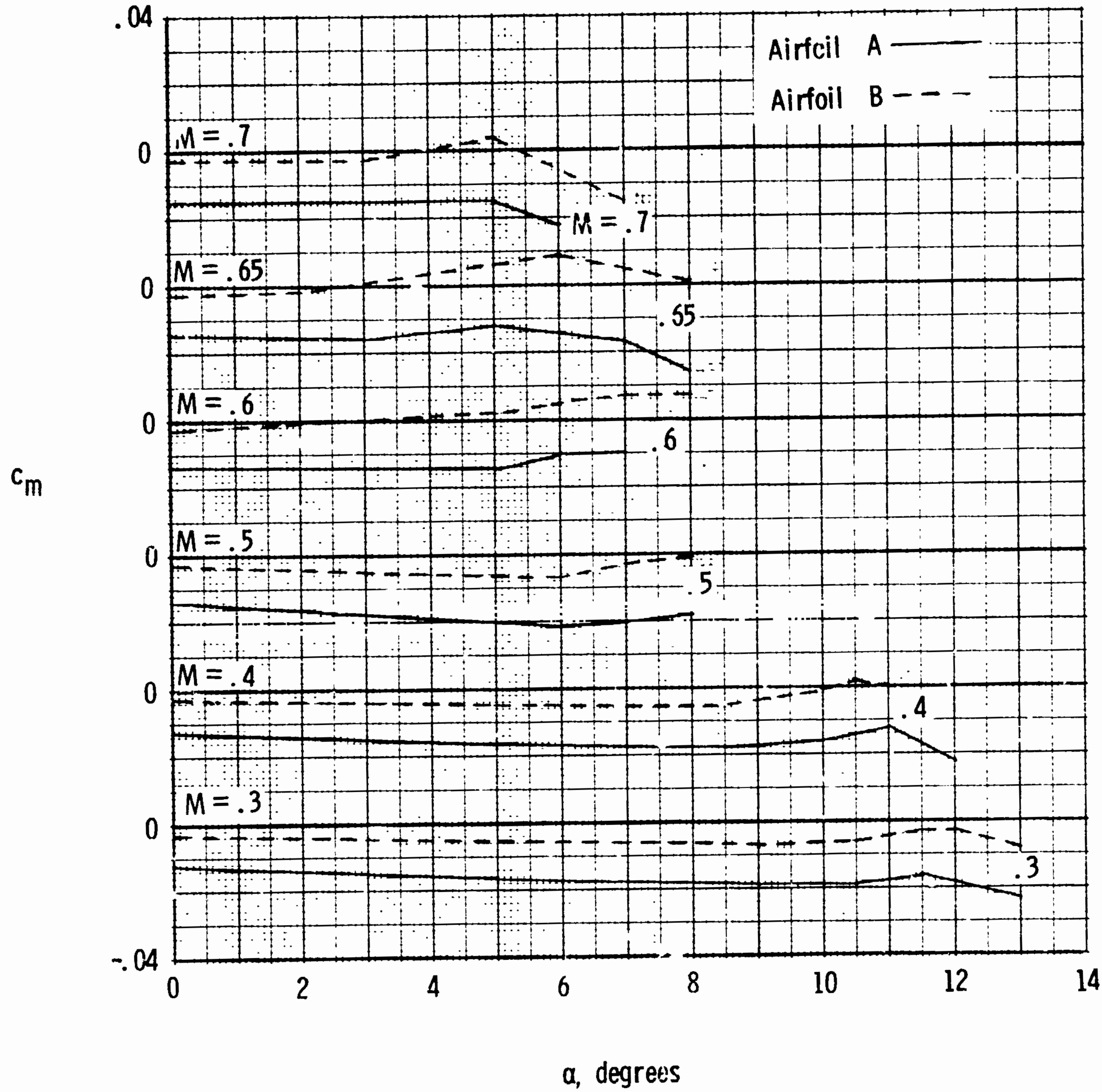

Figure 9.- Predicted pitching-moment coefficient.

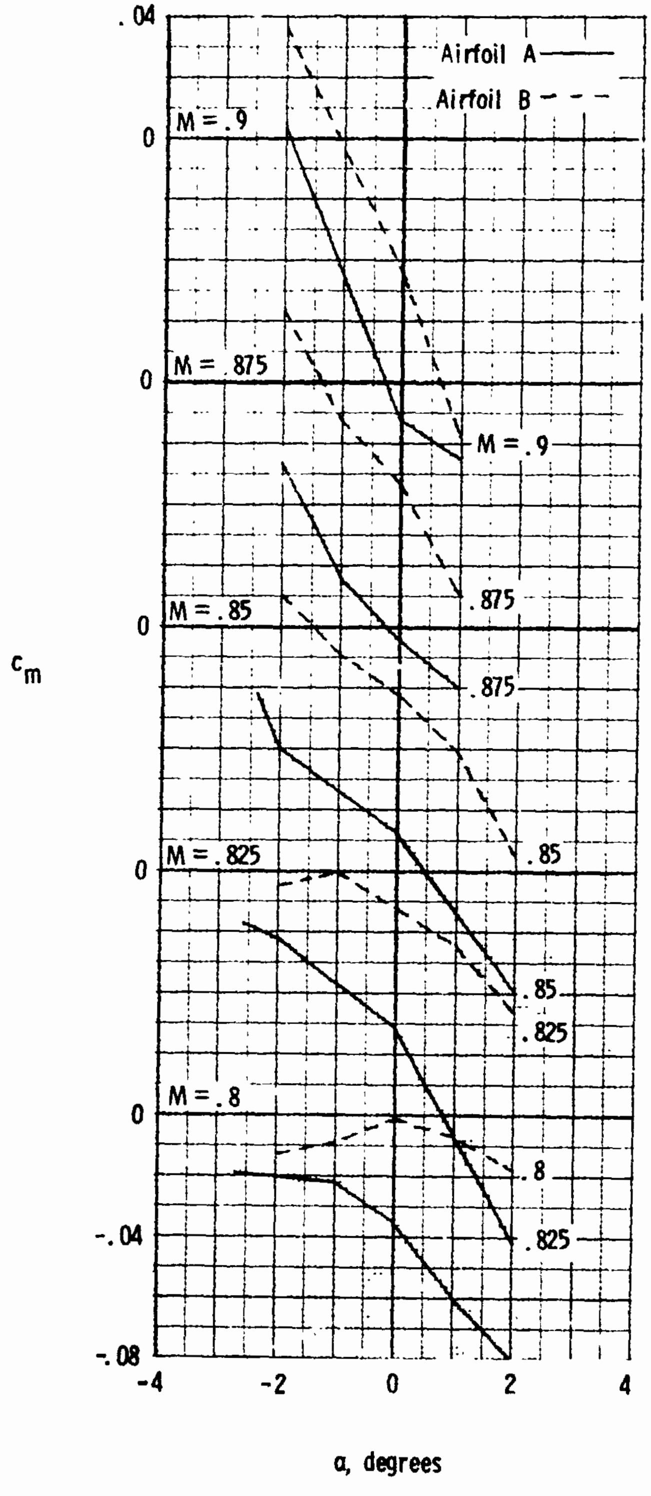

Figure 9.- Concluded.

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15. Supplementary Notes

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16. Abstract

An analytical study has been conducted to predict the aerodynamic characteristics of two helicopter rotor airfoils, versions of the FX69-HL-083 and FX71-H-080. Documentation of the predictive process covers the development of empirical factors used in conjunction with computer programs for airfoil analysis. Tables of lift, drag, and pitching-moment coefficient for each airfoil were prepared for two-dimensional, steady-flow conditions at Mach numbers from 0.3 to 0.9 and Reynolds numbers of 7.7 to 23.0 x 10^6, respectively.

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