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NOMENCLATURE

c	airfoil chord, cm (in.)
c_d	section drag coefficient
c_l	section lift coefficient
c_m	section pitching-moment coefficient referenced to quarter chord
C_p	pressure coefficient $\frac{p_L - p_\infty}{q_\infty}$
h	tunnel height, m (ft)
p	static pressure, N/m^2 (lb/ft^2)
q	dynamic pressure, N/m^2 (lb/ft^2)
Re	Reynolds number based on free-stream conditions and airfoil chord
x	airfoil abscissa, cm (in.)
y	airfoil ordinate, cm (in.)
α	angle of attack, deg

Subscripts:

max	maximum
L	local
∞	free-stream conditions

EFFECTS OF UPPER SURFACE MODIFICATION ON THE AERODYNAMIC CHARACTERISTICS OF THE NACA 63_2-215 AIRFOIL SECTION

Raymond M. Hicks and Edward T. Schairer

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SUMMARY

An upper surface modification designed to increase the maximum lift coefficient of the NACA 63_2-215 airfoil section was tested at Mach numbers of 0.2, 0.3, and 0.4 at Reynolds numbers of 1.3×10^6, 2.0×10^6, and 2.5×10^6 with free transition. The NACA 63_2-215 profile was also tested for comparison with the modified section. The modification increased the thickness over the forward 32% of the upper surface of the airfoil contour.

The modified profile was found to provide substantially higher maximum lift coefficients than the NACA 63_2-215 section at all Mach numbers and Reynolds numbers tested. Relative to the baseline section, the upper-surface modification produced a slightly higher drag level at low and moderate lift coefficients due to less laminar flow on the upper surface. The drag at high lift coefficients was less for the modified section when compared to the 63_2-215 profile due to delayed trailing edge separation. The modified section also showed a small forward shift in the position of the aerodynamic center.

INTRODUCTION

A recent experimental evaluation of a modification to the forward region of the upper surface of the NACA 64_1-212 airfoil (ref. 1) showed that substantial increases in the maximum lift coefficient were achieved at low and moderate Reynolds numbers and Mach numbers. The increased forward thickness of the profile increased the maximum lift coefficient by reducing the peak negative pressures and the adverse pressure gradient near the leading edge, thereby delaying leading edge separation until trailing edge separation is well developed.

The current investigation was conducted to evaluate a similar upper surface modification to a 15% thick NACA 6-series section. The investigation was prompted by a concern that the substantial increase in $c_{l_{max}}$ exhibited by the modified 12% thick, 6-series section would be less for the modified 15% thick 6-series sections because the 15% thick 6-series airfoil exhibited trailing edge stall. Furthermore the type of modification under consideration reduces the adverse pressure gradient over only the forward 20% of the chord leaving the final 80% of the pressure gradient virtually unchanged. However, the results of the investigation, to be discussed later, will show substantial improvement in $c_{l_{max}}$ for the 15% thick 6-series section chosen for this study which indicates that it is important to reduce adverse pressure gradients near the leading edge of trailing edge stall airfoils.

The improvement in $c_{l_{max}}$ shown here and in reference 1 can be achieved without incurring a drag penalty for most airplanes using standard aluminum construction. However, if an airplane wing could be constructed to tight tolerances without protrusions and lap joints aligned normal to the flow direction and if it could be maintained with polished surfaces, the type of modification suggested here would produce a small drag increase due to the loss of some laminar flow over the upper surface of the wing. A test, which clearly demonstrates the difficulty in achieving laminar flow even on a carefully built aluminum wing, was conducted by Beech Aircraft Company in 1945. A preproduction Model 35 Bonanza was flight tested with both NACA 6-series, laminar flow airfoil sections and NACA 5-digit sections. The results of the test showed no difference in cruise speed for the two airfoils but the 5-digit profile proved to be a better airfoil for the entire flight regime because of better low speed characteristics (ref. 2).

AIRFOIL SECTION DESIGN

The NACA 63_2-215 airfoil section and the modified profile are shown in figure 1. The modified section is referred to as Mod. B throughout this report to emphasize the similarity between the modified section of this study and Mod. B section of reference 1. The aerodynamic analysis program[1] used to design the modified profile for the current study is identical to that used during the investigation reported in reference 1. However, the technique used to design the modified profile for the current study was automated by coupling the aerodynamic code to a numerical optimization code. This eliminated the time-consuming procedure used in reference 1 which consisted of several numerical cut-and-file iterations using a large scale drawing of the NACA 64_1-212 profile, a French curve, and a CDC 7600 computer. It is worth noting that the modification for the NACA 64_1-212 section of reference 1 required six manual iterations using about 12 hr of drafting, reading coordinates from drawings, punching the coordinates on computer cards, and organizing the input for the computer; on the contrary, the only time required to develop the modification for the NACA 63_2-215 section of the current study was approximately 15 min of computer input preparation. The numerical optimization technique used during the current study is fully described in reference 3.

In both studies the improvement in $c_{l_{max}}$ was accomplished by reducing the peak negative pressures and adverse pressure gradient near the leading edge of the original 6-series sections. The modification was restricted to the forward section of the profile to minimize the change in both the design lift coefficient and the pitching moment characteristics of the NACA 6-series section. It is important to retain the basic camber of the original airfoil so that existing airplanes can be retrofitted with the type of modification described here without changing the angle of incidence of the wing or requiring additional tail power.

The coordinates of the NACA 63_2-215 and Mod. B sections are given in tables 1 and 2, respectively.

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[1] Transonic Flow Calculations for Airfoils and Bodies of Revolution by Antony Jameson. Grumman Aerodynamics Report, 370-71-1, Dec. 1971.

APPARATUS AND TEST PROCEDURE

Models

Two airfoil models with the NACA 63_2-215 and Mod. B profiles were machined from aluminum billets. Each model has a nominal chord of 20.32 cm (8 in.) and a span of 60.96 cm (24 in.). The models are equipped with 24 upper-surface orifices and 23 lower-surface orifices drilled normal to the surface to determine the pressure distributions on the model surfaces.

Wind Tunnel

The tests were conducted in the Ames 2- by 2-Foot Transonic Wind Tunnel, a variable-speed, continuous flow, ventilated wall, variable pressure facility. The tunnel can be used for two-dimensional testing by replacing the ventilated side walls with solid walls where model-supporting thick glass windows are mounted. The windows can be rotated by a motorized drive system to change the angle of attack. An 82-tube drag rake located 1.31 chords behind the model trailing edge is used to survey the model wake. Figure 2 shows an airfoil model installed in the tunnel along with the drag rake. Airfoil models are mounted spanning the horizontal dimension of the tunnel test section so that the center of rotation of the side windows is near the 25% chord station on the model. The gaps between the ends of the model and side windows were sealed.

Instrumentation

Measurements of the model surface pressures and the wake rake pressures were made by an automatic pressure-scanning system that utilizes precision pressure transducers. Basic tunnel pressures were measured with precision mercury manometers. Angle of attack was measured with a potentiometer operated by the drive gear for the rotating side windows. Data were obtained by a high-speed, data-acquisition system and recorded on paper tape.

Tests

The section aerodynamic characteristics of the two airfoils were obtained at $M = 0.2, 0.3$ and 0.4 at $Re = 1.3 \times 10^6$, 2.0×10^6, and 2.5×10^6. The angles of attack ranged from about -4° to 18°, depending on the stalling angle of each model. The models were tested only with the wake rake installed because previous investigations in the 2- by 2-Foot Wind Tunnel have shown that the effect of the wake rake on the model surface pressures is negligible for the rake position used in the present tests. Data were obtained at all test conditions with free transition since the full scale Reynolds number for most general aviation airplanes flying at approach speed was attained during the test, and because of the difficulty in simulating realistic manufacturing roughness on a wind tunnel model.

Pressure coefficients were determined from surface pressure measurements. Section normal force coefficients, chord force coefficients, and pitching-moment coefficients were obtained from an integration of the pressure coefficients. The pitching-moment coefficients were referenced to the quarter-chord point. Section profile drag was calculated from the wake-rake total and static-pressure measurements.

The model angle of attack was corrected for the presence of the tunnel walls by the following equation:

$$\Delta\alpha = \delta(c/h)c_l$$

where $\Delta\alpha$, δ, c/h, and c_l are the angle-of-attack correction, correction factor, model chord/tunnel height ratio, and section lift coefficient, respectively. The angle-of-attack correction factor δ is a function of Mach number. The following values were used and the corresponding $\Delta\alpha$ was added algebraically to the model geometric angle of attack expressed in degrees:

M	δ
0.2	5.393
.3	8.544
.4	10.593

(The correction factors δ were determined during a tunnel calibration conducted by L. S. Stivers, Jr.) The Mach number corrections due to the presence of the tunnel walls were negligible for the Mach numbers of this investigation.

RESULTS AND DISCUSSION

Lift

The basic force coefficients for the two airfoils tested are presented in figures 3(a) through 3(i). The Mod. B profile gave substantially higher maximum lift coefficient and a somewhat more abrupt stall than the NACA 63_2-215 section. The upper surface modification had little effect on the basic camber distribution of the 63_2-215 airfoil as shown by the similar values of c_l at $\alpha = 0°$ for the two airfoils at all test conditions. As discussed previously, one of the criteria used in developing the upper surface modification was to retain the basic camber distribution of the original 6-series airfoil. This may be important if such modifications are considered for retrofit of existing aircraft.

Summary plots of $c_{l_{max}}$ versus Reynolds number for the three test Mach numbers are presented in figures 4(a) through 4(c). These figures clearly show the higher $c_{l_{max}}$ for the Mod. B profile. The values of $c_{l_{max}}$, shown here may be slightly lower than that achieved in actual use on general aviation airplanes since landing Mach number of most light planes is 0.1 or less and previous

NACA data have shown that $c_{l_{max}}$ can decrease as the Mach number is increased from 0.1 to 0.2 (ref. 4).

Drag

The profile drag data in figure 3 generally show that the drag level of the Mod. B airfoil is somewhat higher than that of the NACA 63_2-215 profile at low and moderate lift coefficients. However, the models used during this study had accurate polished contours and were tested with free transition; hence, the drag data of the 6-series section reflects the existence of more laminar flow on the upper surface than the Mod. B profile. As discussed previously the amount of laminar flow found on the 6-series section tested during this study would be difficult to attain on most in-service aircraft. Hence, the drag difference between the two airfoils shown in figure 3 would not be found with most production airplanes.

Note that the Mod. B profile exhibits lower drag than the 63_2-215 airfoil at high lift coefficients. This drag difference is not due to a difference in the amount of laminar flow present on the models but rather to more separation near the trailing edge of the 63_2-215 profile than for the Mod. B section. Reduced drag at high lift should be of particular interest to the general aviation community because it means lower drag during climb and hence better climb performance, which is important from a safety standpoint.

Pitching Moment

The pitching moment data in figure 3 show that both airfoil sections exhibit similar pitching moment characteristics, with Mod. B giving a slightly more forward aerodynamic center position.

Pressure Distribution

Sample experimental pressure distributions for both airfoil sections are shown in figures 5(a) through 5(d) for a Mach number of 0.2 and a Reynolds number of 2.5×10^6 . Pressure distributions are not shown for all test conditions because of a strong similarity between those shown and the pressure distributions for other test conditions; consequently, the discussion that follows is typical for all test conditions.

Note that the Mod. B profile exhibits a small hump in the pressure distribution and a more adverse pressure gradient than the 63_2-215 section near the leading edge of the upper surface at $\alpha \sim 0°$ and $\alpha \sim 3°$. Such adverse pressure gradients are consistent with the higher drag of the Mod. B section at the low and moderate lift coefficients shown in figure 3. It is evident that such adverse pressure gradients are sufficient to produce transition, which increases the skin friction drag, but are not severe enough to cause separation, as shown by the nearly equal trailing edge pressure recovery for both sections. The pressure distributions for lift coefficients near 1 and 1.5 are shown in figures 5(c) and 5(d), respectively. Note that the pressure peak and adverse pressure gradients are lower for the Mod. B profile than for the 63_2-215 profile at these higher lift coefficients. The smaller adverse pressure gradient near the leading edge of the Mod. B profile at high angles of attack delays trailing edge separation, as shown by comparing the upper surface trailing edge pressure recovery of the two airfoils in figure 5(d). It is clear from figure 5(d) that increasing forward camber of the 63_2-215 section is a good means of delaying stall until higher angles of attack are attained.

When these results are considered along with those of reference 1, it appears reasonable to assume that many 6-series profiles could be converted to good high lift sections by increasing the thickness of the forward region of the upper surface without significantly changing the cruise performance of most production aircraft.

CONCLUSIONS

A wind-tunnel test was conducted to determine the section aerodynamic characteristics of an upper surface modification designed to increase the maximum lift coefficient of the NACA 63_2-215 section. The unmodified 63_2-215 profile was tested for comparison. The test was conducted at $M = 0.2$, 0.3, and 0.4 and at $Re = 1.33 \times 10^6$, 2.0×10^6 and 2.5×10^6. The following results were established.

1. Increasing the upper-surface thickness over the forward 32% of the chord of the NACA 63_2-215 profile increased the maximum lift coefficient of that section from 1.48 to 1.77 at Mach 0.2 and $Re = 2.5 \times 10^6$.

2. The modified profile exhibits similar pitching moment characteristics to the NACA 63_2-215 section with a somewhat more forward aerodynamic center position.

3. The increase in maximum lift coefficient achieved by modifying a 6-series airfoil as described here would not produce a drag penalty for an airplane manufactured with typical aluminum construction.

4. A small drag penalty would be incurred by modifying a 6-series airfoil as described here if the wing were manufactured to tight tolerances without protrusions or joints and maintained in a polished condition while in service.

5. The modified profile exhibits lower drag at high lift coefficients than the 63_2-215 profile which translates into improved climb performance.

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Moffett Field, California 94035, June 9, 1978

REFERENCES

1. Hicks, Raymond M.; Mendoza, Joel P.; and Bandettini Angelo: Effects of Forward Contour Modification on the Aerodynamic Characteristics of the NACA 64_1-212 Airfoil Section. NASA TM X-3293, 1975.

2. Ball, Larry A.: Those Incomparable Bonanzas. McCormick-Armstrong Co., Inc., Wichita, Kansas, 1971.

3. Hicks, Raymond M.; and Vanderplaats, Garret N.: Design of Low Speed Airfoils by Numerical Optimization. Paper 750524 presented at SAE Business Aircraft Meeting, Wichita, Kansas, 1975.

4. Racisz, Stanley F.: Effects of Independent Variation of Mach Number and Reynolds Number on the Maximum Lift Coefficients of Four NACA 6-Series Airfoil Sections. NACA TN 2824, 1952.

TABLE 1.– NACA 63_2-215 AIRFOIL COORDINATES

Upper surface		Lower surface		Upper surface		Lower surface	
x/c	y/c	x/c	y/c	x/c	y/c	x/c	y/c
0	0	0	0	0.3750	0.0852	0.3750	-0.0641
.0002	.0040	.0002	-.0011	.4000	.0845	.4000	-.0632
.0004	.0050	.0004	-.0022	.4250	.0835	.4250	-.0618
.0006	.0057	.0006	-.0031	.4500	.0819	.4500	-.0601
.0008	.0064	.0008	-.0037	.4750	.0800	.4750	-.0580
.0010	.0070	.0010	-.0042	.5000	.0777	.5000	-.0556
.0020	.0092	.0020	-.0063	.5250	.0750	.5250	-.0530
.0030	.0110	.0030	-.0079	.5500	.0720	.5500	-.0501
.0040	.0125	.0040	-.0093	.5750	.0688	.5750	-.0470
.0050	.0138	.0050	-.0104	.6000	.0653	.6000	-.0438
.0100	.0187	.0100	-.0150	.6250	.0615	.6250	-.0404
.0200	.0258	.0200	-.0211	.6500	.0576	.6500	-.0368
.0300	.0313	.0300	-.0256	.6750	.0534	.6750	-.0332
.0400	.0361	.0400	-.0294	.7000	.0491	.7000	-.0295
.0500	.0403	.0500	-.0328	.7250	.0447	.7250	-.0258
.0600	.0440	.0600	-.0357	.7500	.0402	.7500	-.0221
.0700	.0475	.0700	-.0384	.7750	.0357	.7750	-.0185
.0800	.0506	.0800	-.0408	.8000	.0311	.8000	-.0150
.0900	.0535	.0900	-.0430	.8250	.0266	.8250	-.0117
.1000	.0561	.1000	-.0450	.8500	.0222	.8500	-.0086
.1250	.0621	.1250	-.0494	.8750	.0179	.8750	-.0058
.1500	.0671	.1500	-.0531	.9000	.0137	.9000	-.0033
.1750	.0714	.1750	-.0562	.9250	.0098	.9250	-.0013
.2000	.0751	.2000	-.0588	.9500	.0062	.9500	.0001
.2250	.0781	.2250	-.0609	.9600	.0048	.9600	.0005
.2500	.0806	.2500	-.0625	.9700	.0036	.9700	.0007
.2750	.0825	.2750	-.0637	.9800	.0023	.9800	.0008
.3000	.0840	.3000	-.0645	.9900	.0012	.9900	.0006
.3250	.0849	.3250	-.0648	.9950	.0006	.9950	.0003
.3500	.0853	.3500	-.0647	1.[illegible]	0.[illegible]	1[illegible]	0.[illegible]

TABLE 2.– MOD. B AIRFOIL COORDINATES

Upper surface		Lower surface		Upper surface		Lower surface	
x/c	y/c	x/c	y/c	x/c	y/c	x/c	y/c
0	0	0	0	0.3750	0.0852	0.3750	-0.0641
.0002	.0034	.0002	-.0011	.4000	.0845	.4000	-.0632
.0004	.0051	.0004	-.0022	.4250	.0835	.4250	-.0618
.0006	.0064	.0006	-.0031	.4500	.0819	.4500	-.0601
.0008	.0075	.0008	-.0037	.4750	.0800	.4750	-.0580
.0010	.0085	.0010	-.0042	.5000	.0777	.5000	-.0556
.0020	.0125	.0020	-.0063	.5250	.0750	.5250	-.0530
.0030	.0156	.0030	-.0079	.5500	.0720	.5500	-.0501
.0040	.0182	.0040	-.0093	.5750	.0688	.5750	-.0470
.0050	.0205	.0050	-.0104	.6000	.0653	.6000	-.0438
.0100	.0293	.0100	-.0150	.6250	.0615	.6250	-.0404
.0200	.0408	.0200	-.0211	.6500	.0576	.6500	-.0368
.0300	.0489	.0300	-.0256	.6750	.0534	.6750	-.0332
.0400	.0550	.0400	-.0294	.7000	.0491	.7000	-.0295
.0500	.0599	.0500	-.0328	.7250	.0447	.7250	-.0258
.0600	.0640	.0600	-.0357	.7500	.0402	.7500	-.0221
.0700	.0673	.0700	-.0384	.7750	.0357	.7750	-.0185
.0800	.0702	.0800	-.0408	.8000	.0311	.8000	-.0150
.0900	.0727	.0900	-.0430	.8250	.0266	.8250	-.0117
.1000	.0748	.1000	-.0450	.8500	.0222	.8500	-.0086
.1250	.0788	.1250	-.0494	.8750	.0179	.8750	-.0058
.1500	.0816	.1500	-.0531	.9000	.0137	.9000	-.0033
.1750	.0835	.1750	-.0562	.9250	.0098	.9250	-.0013
.2000	.0847	.2000	-.0588	.9500	.0062	.9500	.0001
.2250	.0855	.2250	-.0609	.9600	.0048	.9600	.0005
.2500	.0859	.2500	-.0625	.9700	.0036	.9700	.0007
.2750	.0860	.2750	-.0637	.9800	.0023	.9800	.0008
.3000	.0859	.3000	-.0645	.9900	.0012	.9900	.0006
.3250	.0856	.3250	-.0648	.9950	.0006	.9950	.0003
.3500	.0853	.3500	-.0647	1.[illegible]	0[illegible]	1[illegible]	0.[illegible]

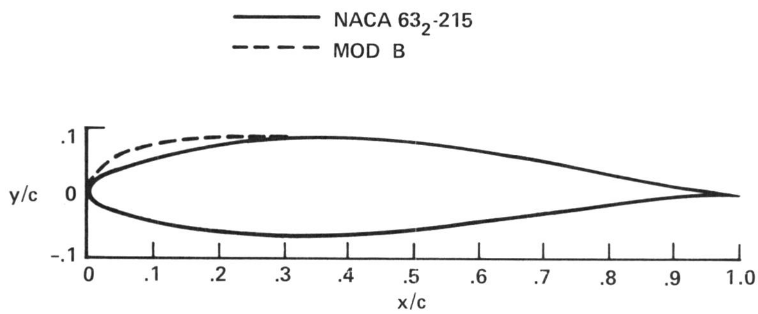


Figure 1.– Airfoil sections tested.

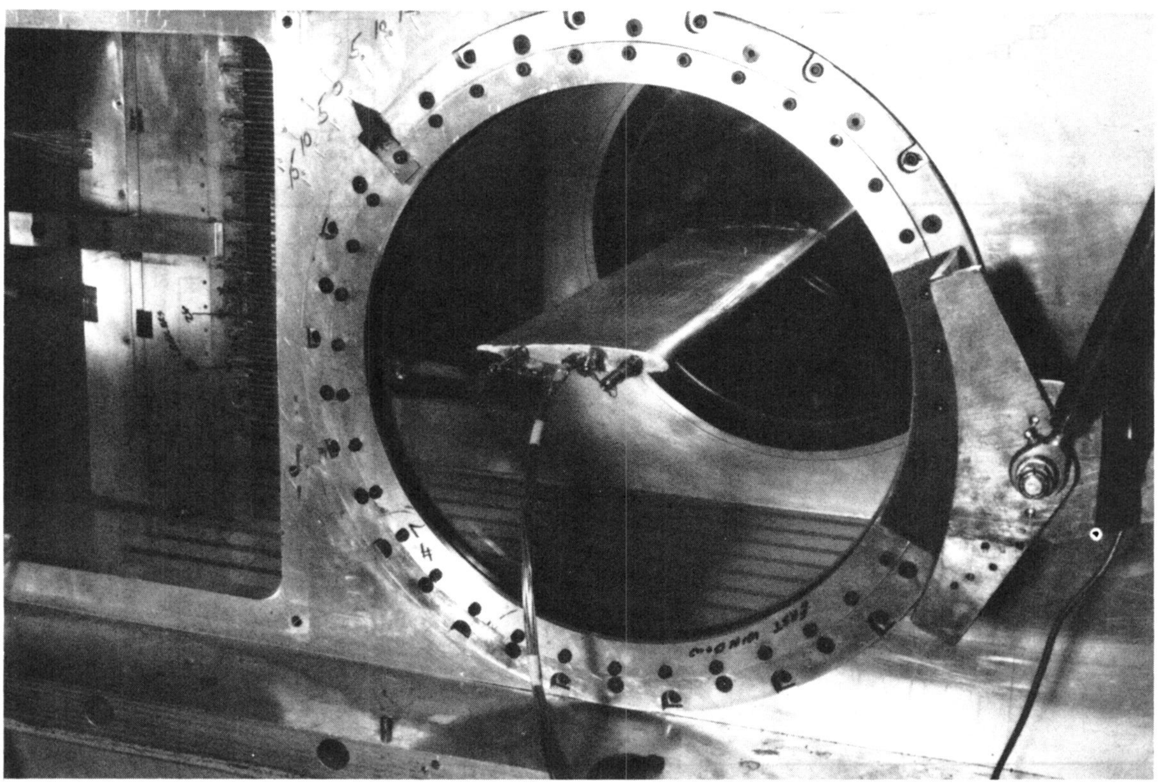
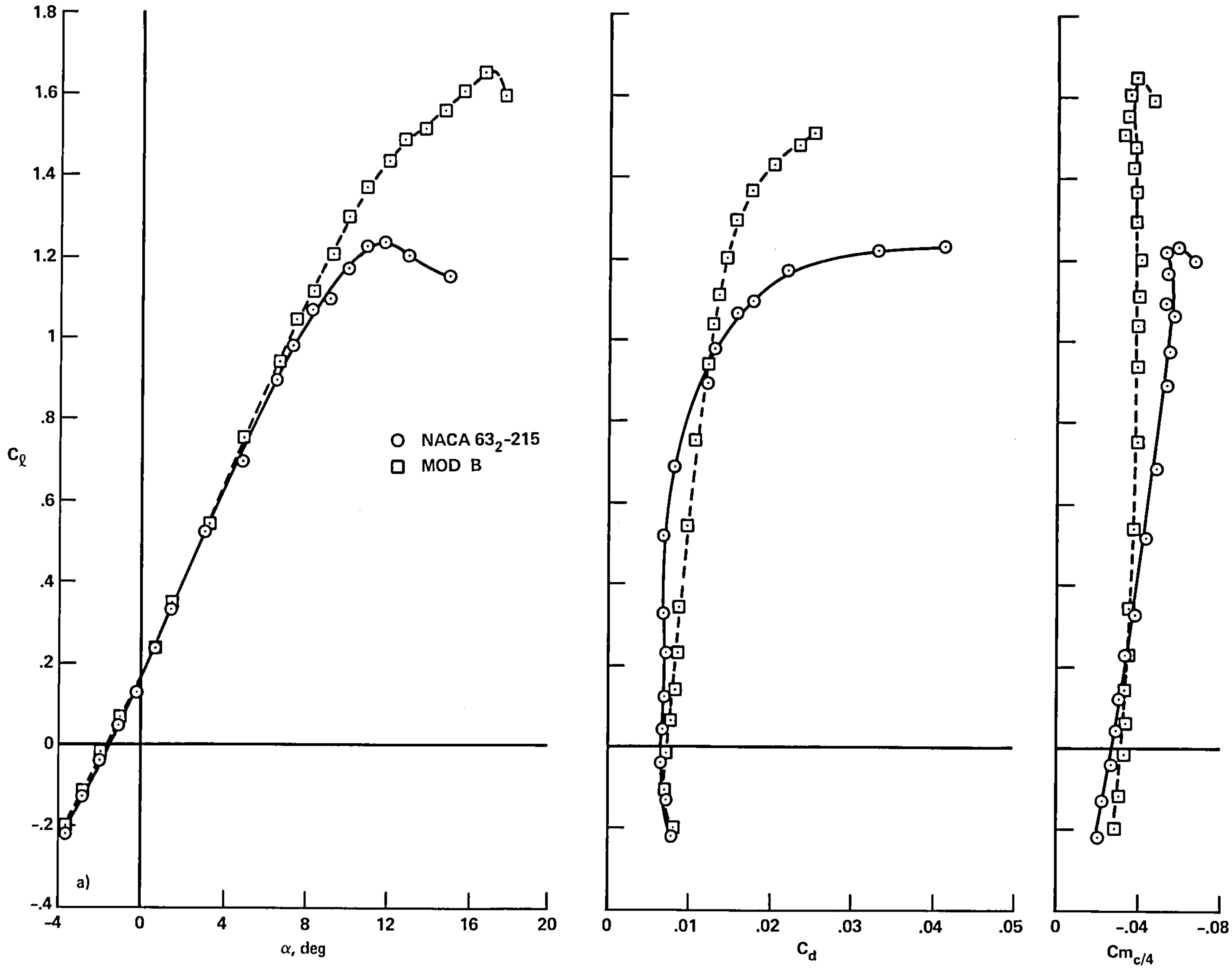
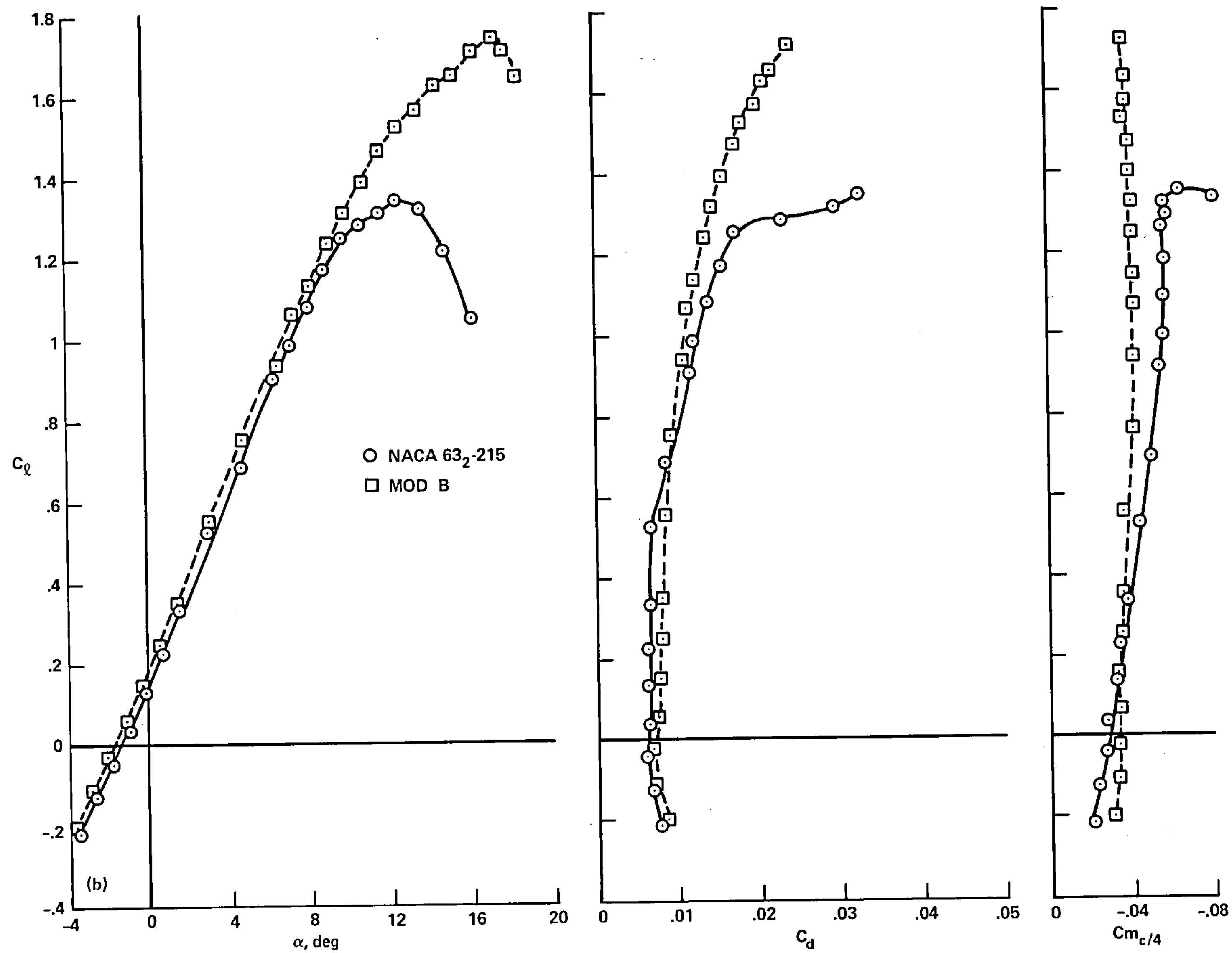


Figure 2.– Airfoil model installed in the Ames 2- by 2-Foot Wind Tunnel.



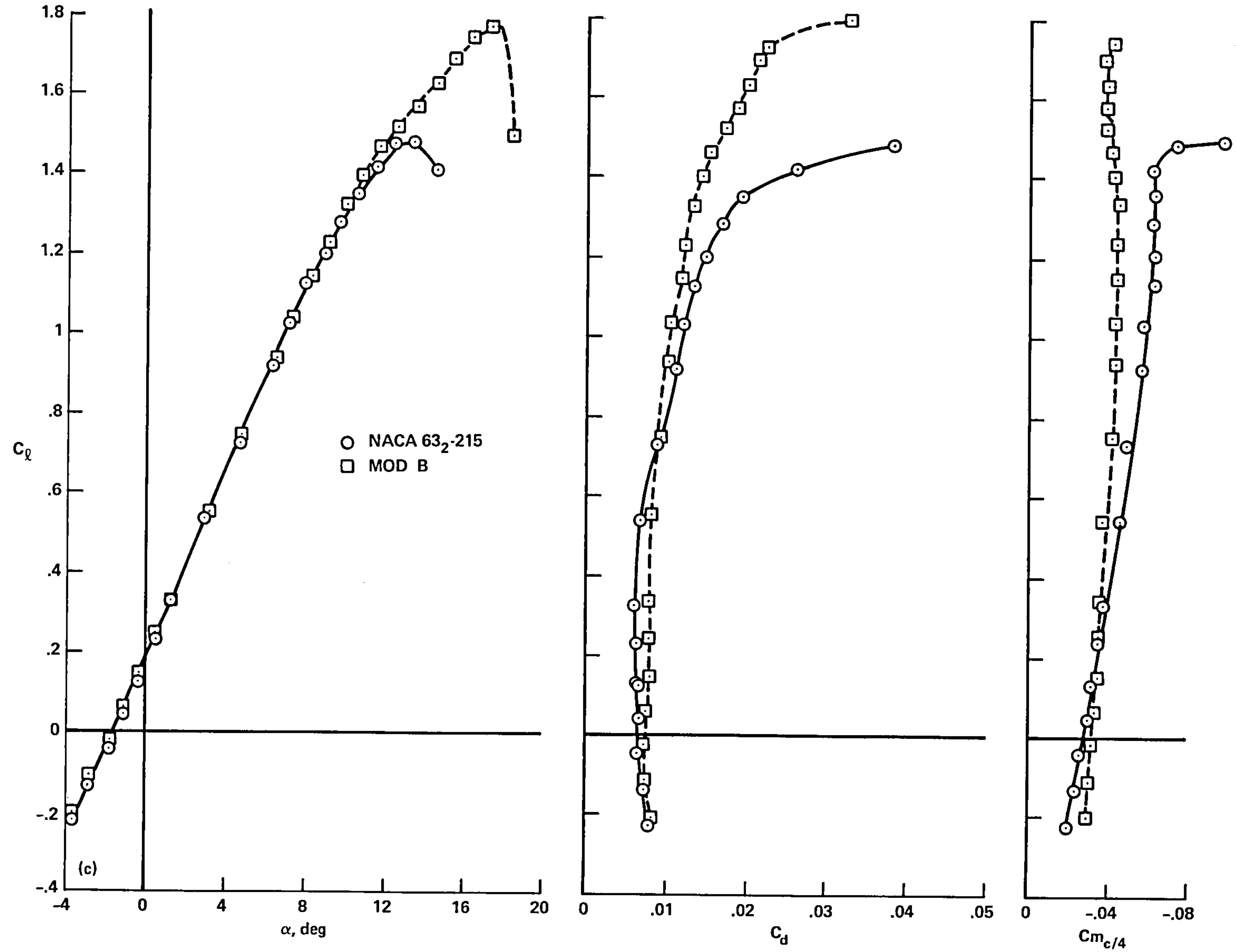
(a) M = 0.2; Re = 1.3×10^6

Figure 3.– Effect of airfoil contour modification on section characteristics, free transition.



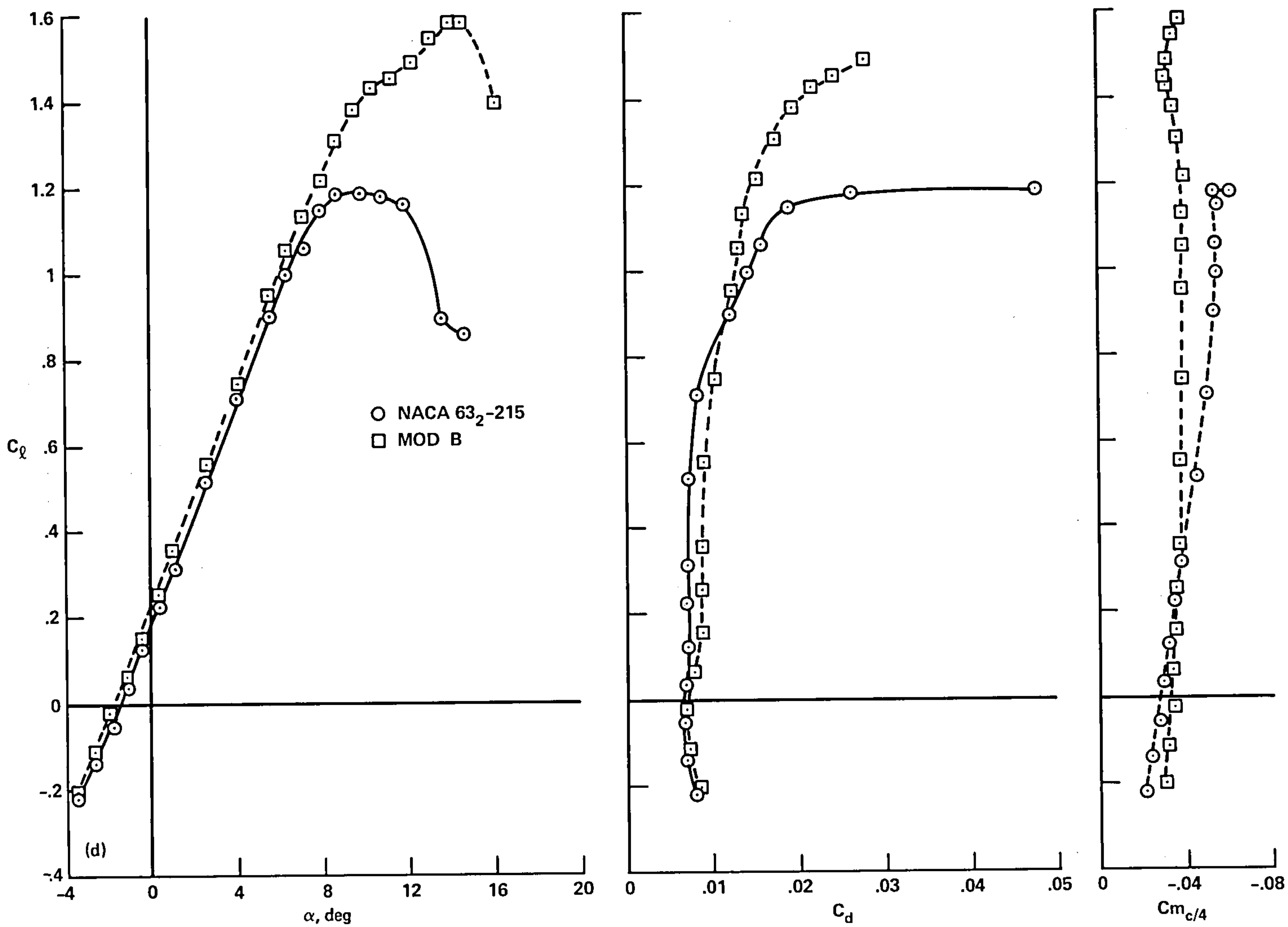
(b) M = 0.2; Re = 2X10[6]

Figure 3.– Continued.



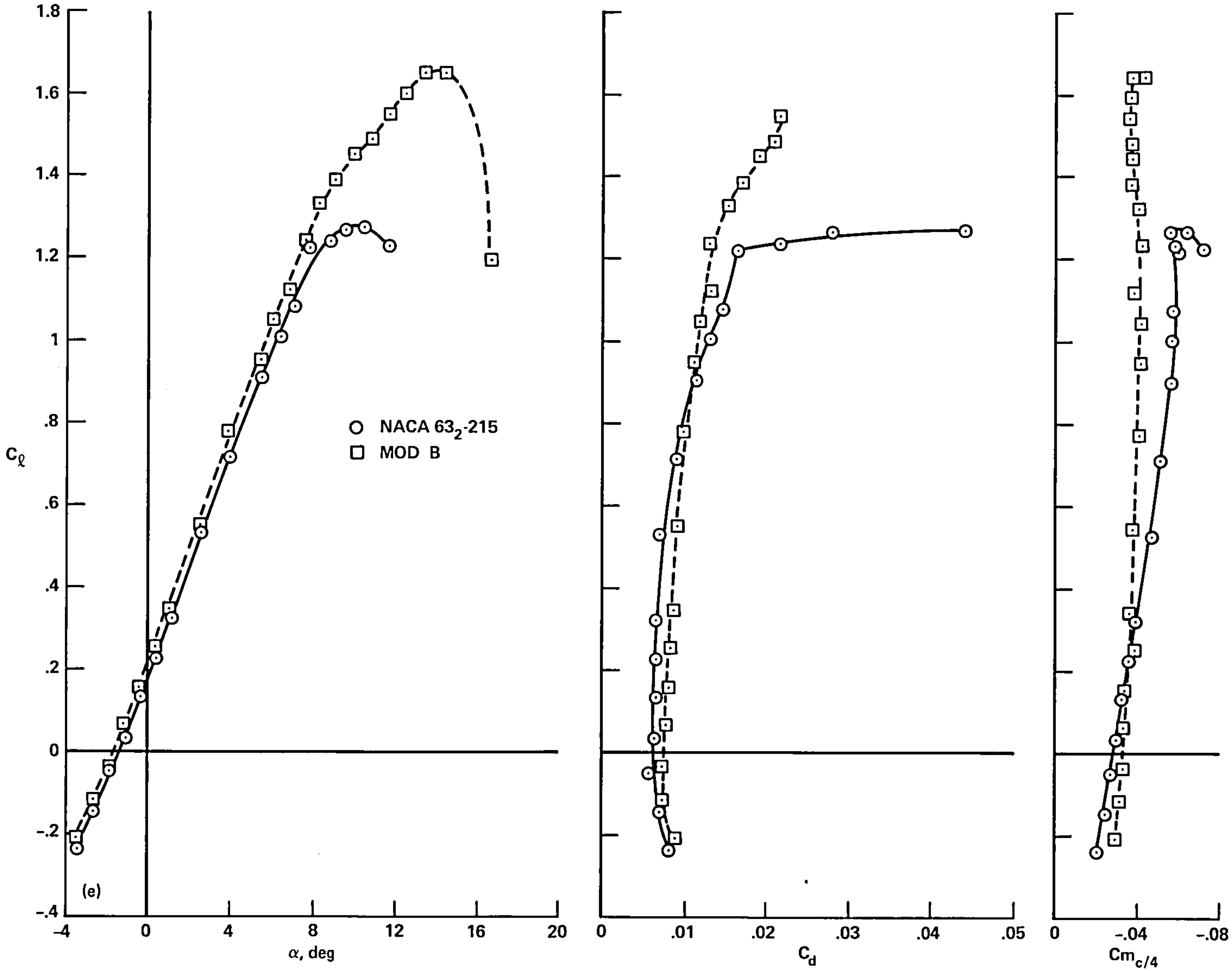
(c) M = 0.2; Re = 2.5×10^6

Figure 3.– Continued.



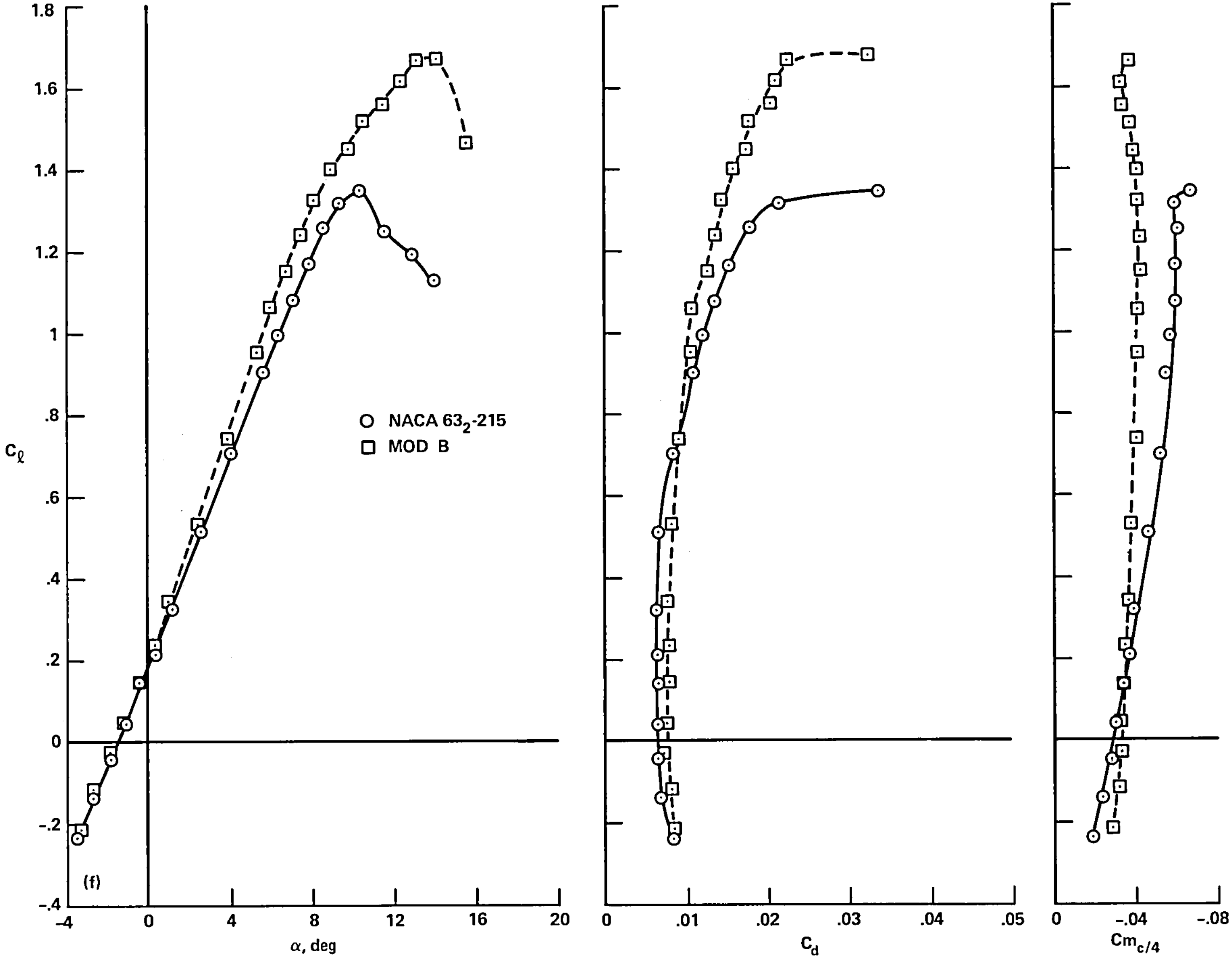
(d) M = 0.3; Re = 1.36X10^6

Figure 3.– Continued.



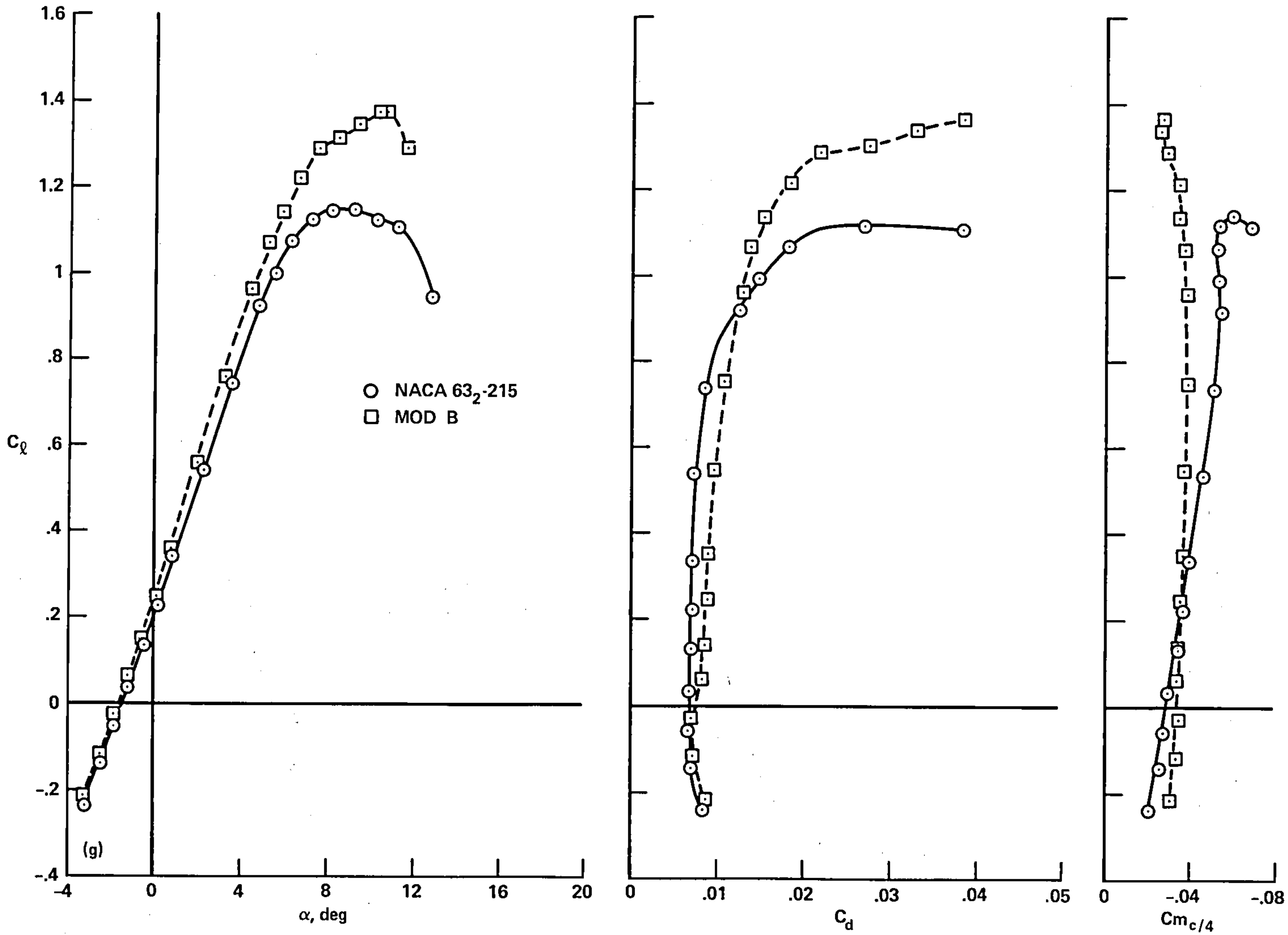
(e) M = 0.3; Re = 2×10^6

Figure 3.– Continued.



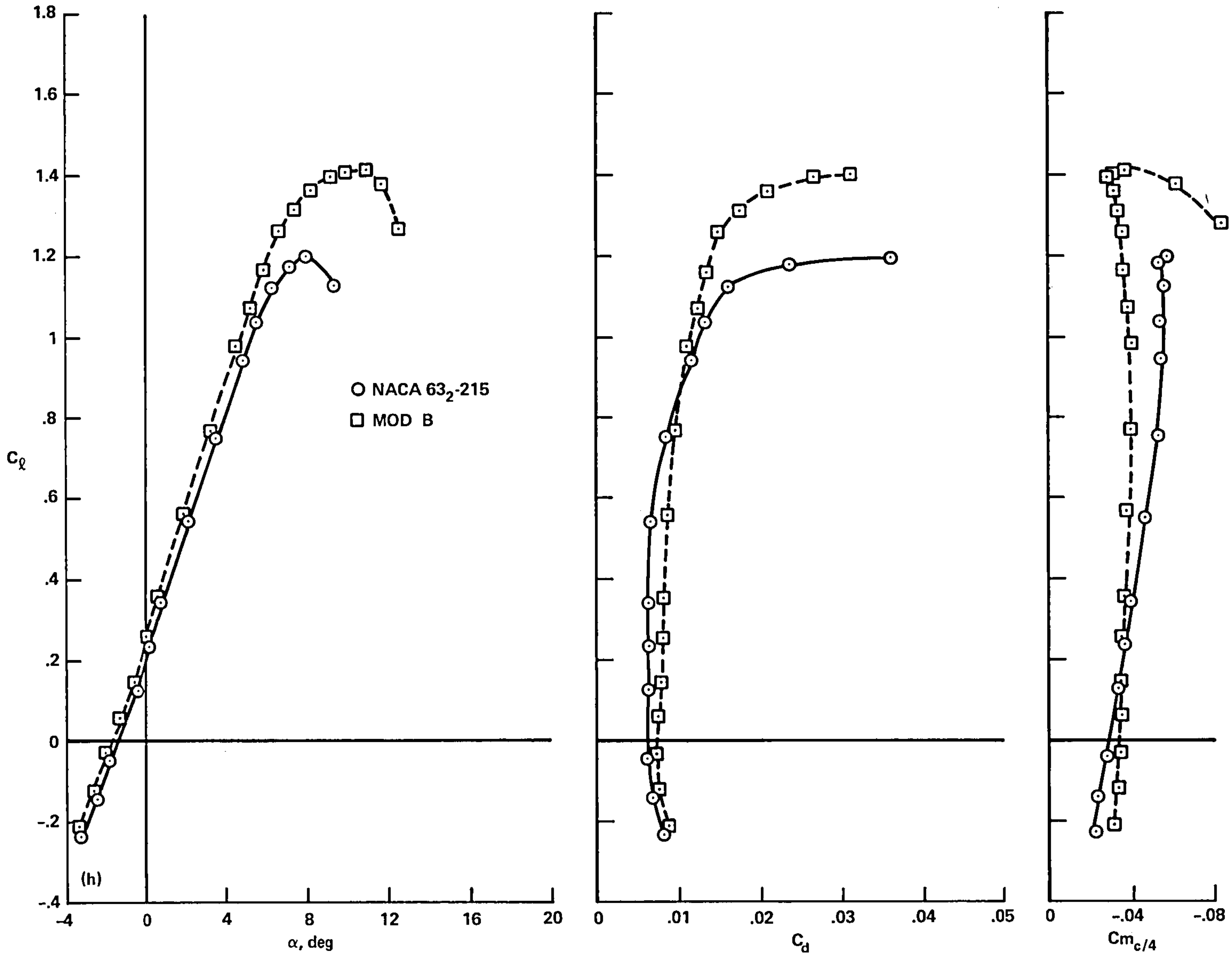
(f) M = 0.3; Re = 2.5X10^6

Figure 3.– Continued.



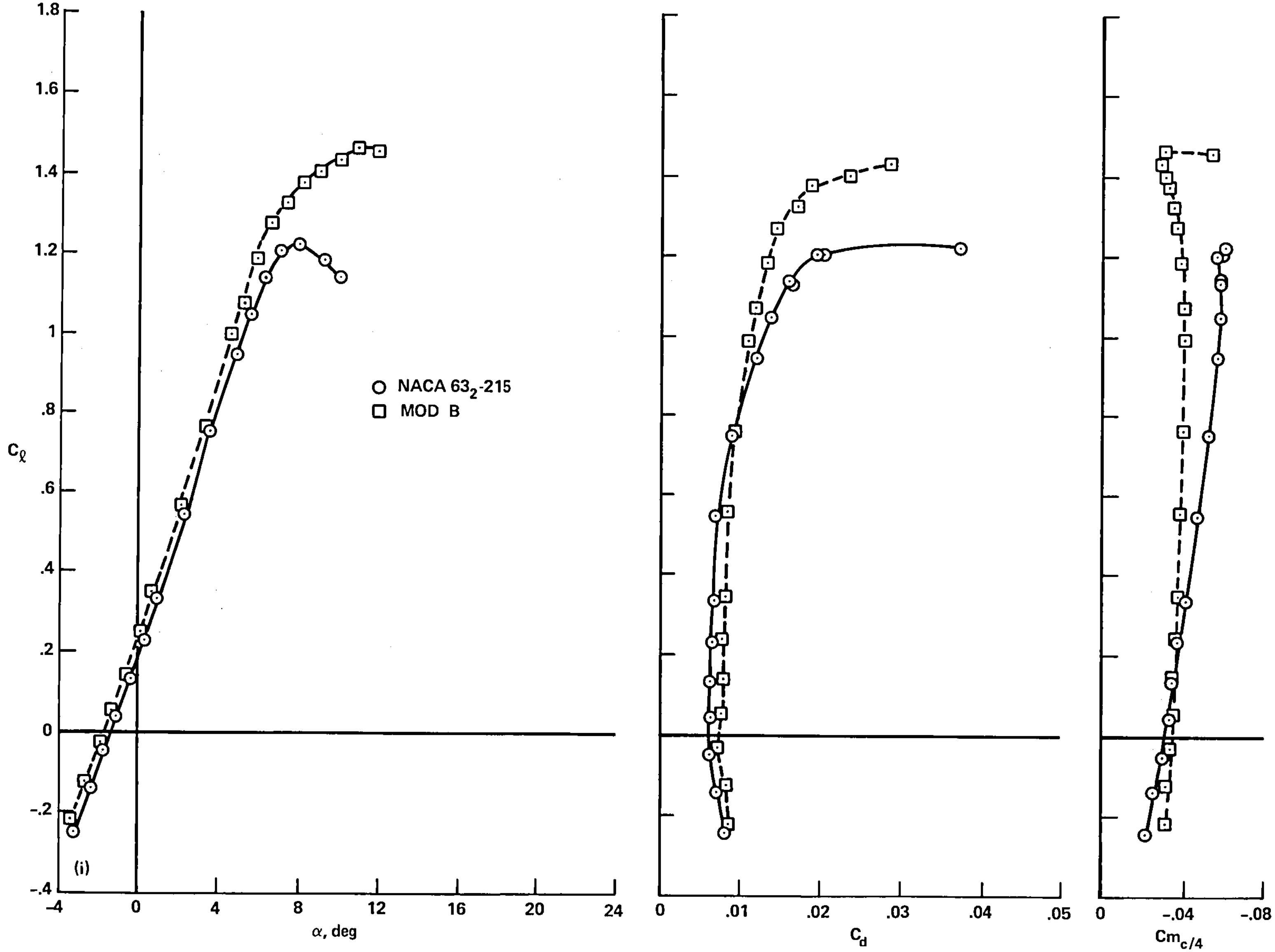
(g) M = 0.4; Re = 1.33X10^6

Figure 3.– Continued.



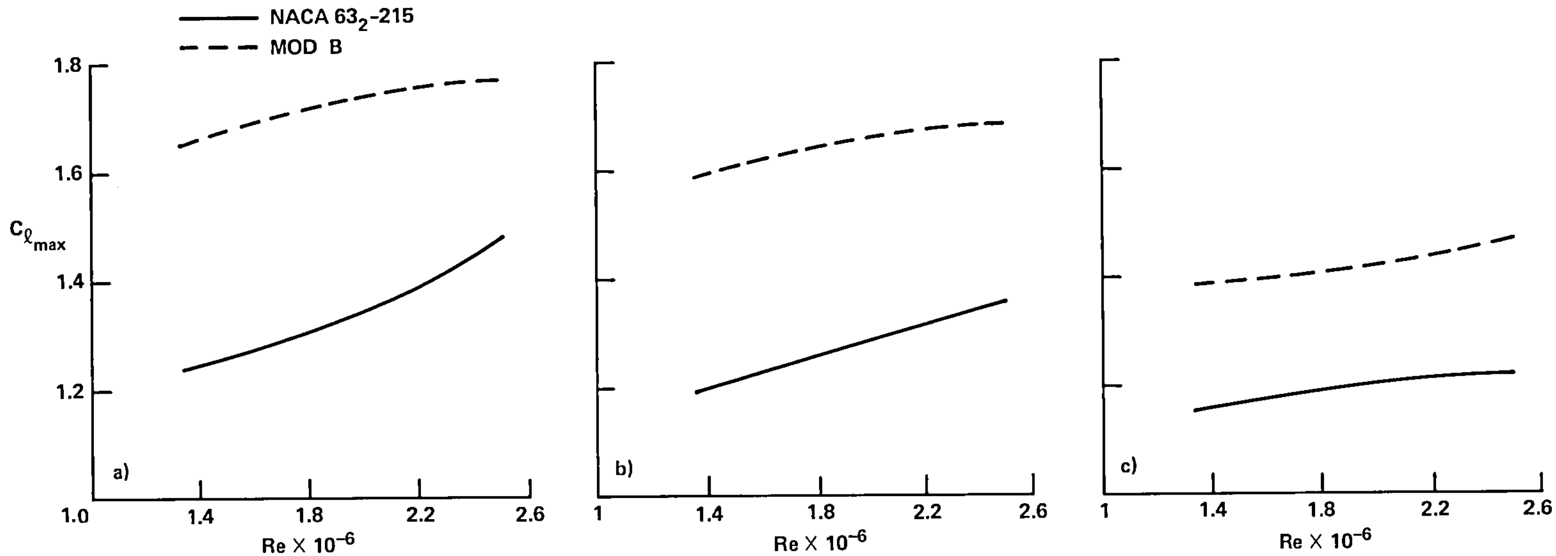
(h) M = 0.4; Re = 2×10^6

Figure 3.– Continued.



(i) M = 0.4; Re = 2.5×10^6

Figure 3.– Concluded.

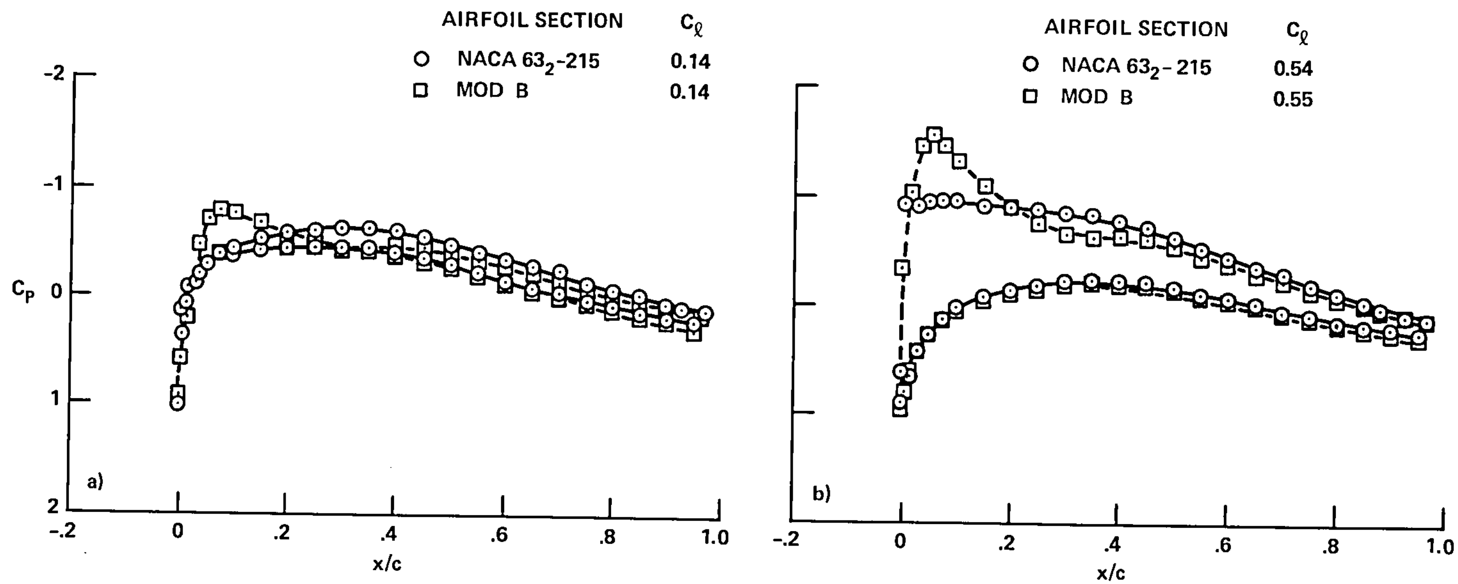


(a) M = 0.2.

(b) M = 0.3

(c) M = 0.4

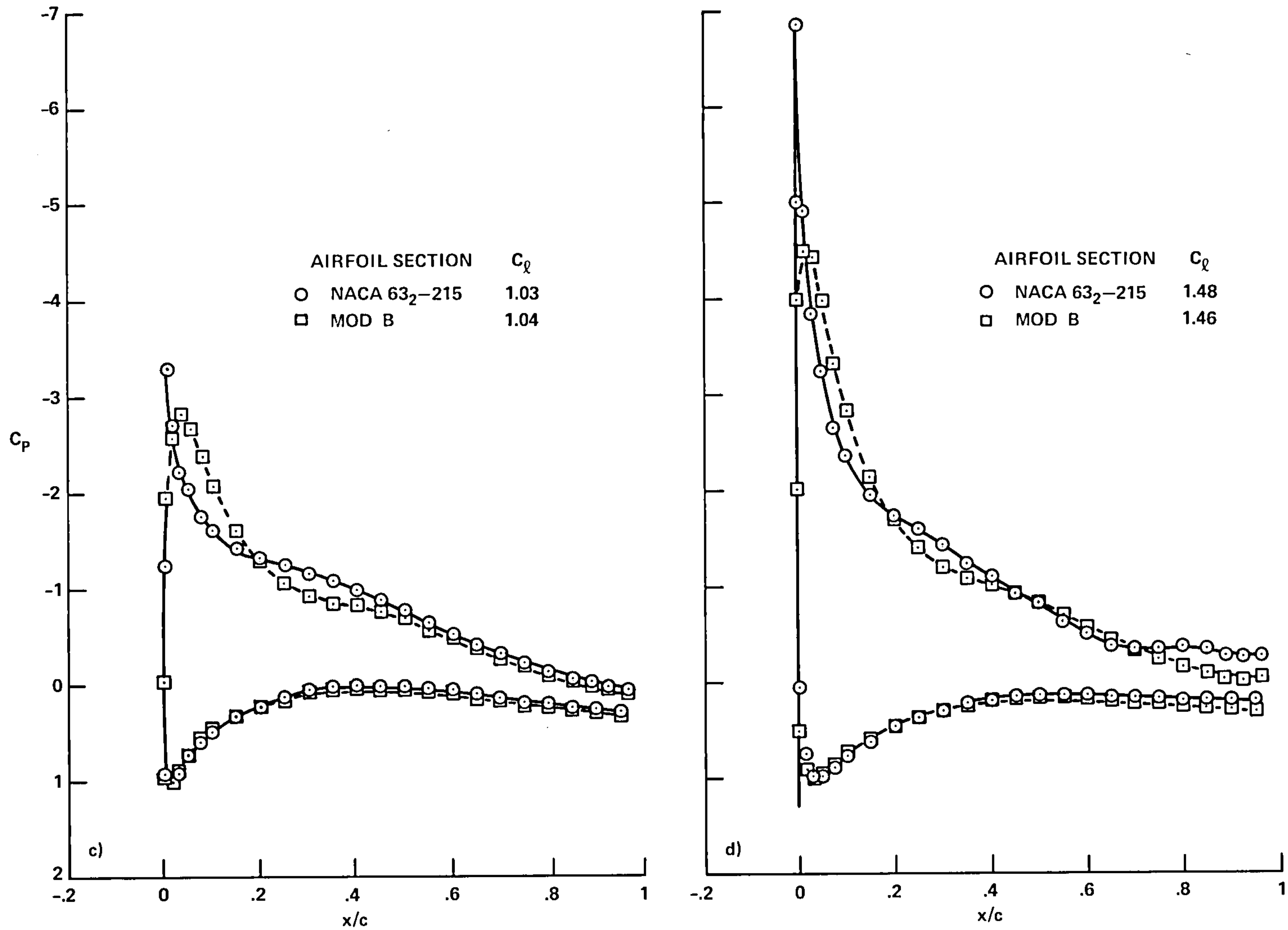
Figure 4.– Effect of airfoil contour modification on maximum lift coefficient, free transition.



(a) $\alpha \sim 0°$

(b) $\alpha \sim 3°$

Figure 5.– Experimental pressure distributions; M = 0.2, Re = 2.5×10^6.



(c) $\alpha \sim 7°$

(d) $\alpha \sim 12°$

Figure 5.– Concluded.

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7. Author(s) Raymond M. Hicks and Edward T. Schairer		8. Performing Organization Report No. A-7507
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		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

An upper surface modification designed to increase the maximum lift coefficient of the NACA 63_2-215 airfoil section was tested at Mach numbers of 0.2, 0.3, and 0.4 at Reynolds numbers of 1.3×10^6, 2×10^6, and 2.5×10^6. Comparisons of the NACA 63_2-215 airfoil aerodynamic coefficients before and after the modification was made. The upper surface modification increased the maximum lift coefficient of the 63_2-215 airfoil significantly at all conditions (e.g., from 1.48 to 1.77 at Mach 0.2 and a Reynolds number of 2.5×10^6).

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