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MEMORANDUM

CHORDWISE PRESSURE DISTRIBUTIONS OVER SEVERAL NACA 16-SERIES AIRFOILS AT TRANSONIC MACH NUMBERS UP TO 1.25

By Charles L. Ladson

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON
June 1959

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CHORDWISE PRESSURE DISTRIBUTIONS OVER SEVERAL NACA 16-SERIES AIRFOILS AT TRANSONIC MACH NUMBERS UP TO 1.25

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SUMMARY

A two-dimensional wind-tunnel investigation of the pressure distributions over several NACA 16-series airfoils with thicknesses of 4, 6, 9, and 12 percent of the chord and design lift coefficients of 0, 0.2, and 0.5 has been conducted in the Langley airfoil test apparatus at transonic Mach numbers from 0.7 to 1.25. The tests ranged in Reynolds number from 2.4×10^6 to 2.8×10^6 and in angle of attack from -10° to 12°. Chordwise pressure distributions and schlieren flow photographs are presented without analysis.

INTRODUCTION

The rotational speeds of helicopter blades are being increased in an effort to provide helicopters with higher forward flight speeds. As a result of this, some consideration is being given to combinations of rotational speed and forward speed that will involve operating local blade elements in the transonic and low supersonic speed regime. To predict loads on these blades, two-dimensional airfoil data are needed in this speed range and the NACA 16-series airfoils should provide useful data for this purpose. Existing data on NACA 16-series airfoils are generally limited to a Mach number of about 0.8 (for example, see refs. 1 and 2). Therefore an investigation was undertaken in the Langley airfoil test apparatus to extend the data on thin and moderately thick 16-series airfoils incorporating camber throughout the transonic speed regime.

The investigation was made on several NACA 16-series airfoils with thickness ratios of 4, 6, 9, and 12 percent and design lift coefficients of 0, 0.2, and 0.5 over a Mach number range from 0.7 to 1.25. The corresponding Reynolds number of the 4-inch-chord models tested at a stagnation pressure of 26 pounds per square inch absolute varied from 2.4×10^6 to 2.8×10^6. The cambered models were varied in angle of attack from -10° to 12° while the symmetrical models were varied from

0° to 12°. Chordwise pressure distributions and schlieren flow photographs were obtained and are presented without analysis.

APPARATUS

The tests were conducted in the Langley airfoil test apparatus (ATA) which is a two-dimensional slotted-throat facility operating on direct blowdown from a supply of dry compressed air. (See fig. 1(a).) The facility incorporates mechanical features which permit independent control of both stagnation pressure and free-stream Mach number. The settling-chamber stagnation pressure is controlled by a pneumatic pressure-regulating valve which enables tests to be made at any constant stagnation pressure from 26 to 60 pounds per square inch absolute.

Air enters the 4- by 19-inch slotted test section through a sonic nozzle from a circular settling chamber about 5 feet in diameter. The area contraction ratio from settling chamber to test section is about 45:1. Three longitudinal slots are located in each of the 4-inch-wide walls, the slots having a total width of 1/2 inch or 1/8 open area. As seen in figure 1(a), the slots begin (at tunnel station 45) 25 inches upstream of the test-region center line and extend slightly downstream of the test region. Figure 1(b) presents a more detailed sketch of the test section, showing the plenum chamber which surrounds the 4- by 19-inch test region. Ducts (see fig. 1(b)) of 62-square-inch cross section connect the two plenum chambers adjacent to each of the slotted walls to eliminate any pressure differentials which may have existed. Air which has passed through the slots into the plenum chamber is returned to the main airstream over reentrant flow fairings downstream of the test section. The minimum area in this mixing section ahead of the chokers is 20 percent larger than the test-section area to provide space for the low-energy reentrant flow to return to the main stream. Tunnel calibrations showed that the 20-percent increase in area limited the maximum test Mach number to 1.25.

A choker section located downstream of the mixing region (between tunnel stations 98 and 120) is used to control the Mach number. The two 19-inch-high side walls are made of thin flexible metal so that they may be deflected into the airstream. When deflected, these walls decrease the cross-sectional area where sonic speed is maintained and thus decrease the tunnel mass flow. The test-section Mach number can be set at any value, depending upon the amount of deflection of these flexible walls.

A transition section is located downstream of the choker section and is followed by a conical diffuser which exhausts the tunnel to the atmosphere.

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An examination of the tunnel-empty static-pressure measurements shows that the Mach number variation from 1 chord length ahead of to 1 chord length behind the center of the test region (station 70) is about ±0.002. A similar examination of the Mach number gradients in the normal-to-stream direction (taken at tunnel station 70) shows a maximum variation of ±0.010 from 1 chord length above to 1 chord length below the model chord line.

MODELS

The models tested in this investigation were the NACA 16-004, 16-006, 16-009, 16-012, 16-206, 16-506, and 16-512 airfoil sections. Ordinates for the NACA 16-series airfoils are presented in table I, which is taken from reference 2. All models were of 4-inch chord and completely spanned the 4-inch width of the tunnel. Static-pressure orifices having diameters of 0.0135 inch (No. 80 drill) were drilled normal to the airfoil surface near the midspan section. All models had orifices located at the 2.5-, 5.0-, 7.5-, 10-, 15-, 20-, 25-, 30-, 35-, 40-, 45-, 50-, 55-, 60-, 65-, 70-, 75-, 80-, 85-, and 90-percent-chord stations on both upper and lower surfaces; several of the thicker models had additional orifices at the 1.25- and 95-percent-chord stations.

TESTS

Methods and Range

The orifices in the model were connected to a multitube mercury manometer and the results were photographed during the run. From these photographs the pressure distributions presented herein were plotted. Schlieren motion pictures of the flow past the models were obtained during separate tests, and representative frames have been presented. The exposure time of each frame is approximately 4 microseconds.

The tests covered a Mach number range from 0.7 to 1.25, an angle-of-attack range from -10° to 12°, and a Reynolds number range of 2.4×10^6 to 2.8×10^6. The stagnation pressure was held constant throughout the tests at 26 pounds per square inch absolute.

Corrections

Force coefficients from the airfoil test apparatus on an NACA 0012 airfoil section are compared with data from the Langley 4- by 19-inch semiopen (converted to blowdown operation) tunnel in figure 2. The calibration of the 4- by 19-inch semiopen tunnel is presented in reference 3;

however, a settling chamber has been installed at the entrance of the tunnel to convert it to direct blowdown operation and to eliminate water-vapor condensation effects. From figure 2 it is seen that the data from the two facilities are in fairly close agreement.

The major correction to which the data from the two facilities are subject is a correction to angle of attack. The correction for the 4- by 19-inch semiopen tunnel has been evaluated by the method of reference 4 and is presented in reference 3. The theoretical value of the correction to the data from the 1/8-open slotted-wall ATA is about twice that derived for the 4- by 19-inch semiopen tunnel. However, the data of figure 2 indicate that the correction at the lower speeds should be about the same for the two facilities. Since no reliable corrections are currently available, the data presented herein are uncorrected.

RESULTS

The results of this investigation are presented as faired pressure distributions and schlieren flow photographs without analysis. The plotted test points in the pressure distributions that are neglected in the fairing (as in figure 5) are known to be faulty. An index presenting the figure numbers for the various models and angles of attack is presented in table II.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., March 9, 1959.

REFERENCES

1. Stack, John: Tests of Airfoils Designed to Delay the Compressibility Burble. NACA Rep. 763, 1943. (Supersedes NACA TN 976.)

2. Lindsey, W. F., Stevenson, D. B., and Daley, Bernard N.: Aerodynamic Characteristics of 24 NACA 16-Series Airfoils at Mach Numbers Between 0.3 and 0.8. NACA TN 1546, 1948.

3. Daley, Bernard N., and Dick, Richard S.: Effect of Thickness, Camber, and Thickness Distribution on Airfoil Characteristics at Mach Numbers up to 1.0. NACA TN 3607, 1956. (Supersedes NACA RM L52G31a.)

4. Katzoff, S., Gardner, Clifford S., Diesendruck, Leo, and Eisenstadt, Bertram J.: Linear Theory of Boundary Effects in Open Wind Tunnels With Finite Jet Lengths. NACA Rep. 976, 1950. (Supersedes NACA TN 1826.)

TABLE I.- THICKNESS DISTRIBUTION AND MEAN-LINE CHARACTERISTICS FOR NACA 16-SERIES AIRFOILS

[All linear values in percent chord]

Station	NACA 16-009 ordinate (a)	Mean-line characteristics for design lift coefficient of 1.0 (b)	
		Ordinate	Slope
0	0	0	0.62234
.6	.676	.295	.40665
1.25	.969	.535	.34771
2.5	1.354	.930	.29155
5	1.882	1.580	.23432
7.5	2.274	2.120	.19993
10	2.593	2.587	.17486
15	3.101	3.364	.13804
20	3.498	3.982	.11032
25	3.812	4.475	.08743
30	4.063	4.861	.06743
40	4.391	5.356	.03227
50	4.500	5.516	0
60	4.376	5.356	-.03227
70	3.952	4.861	-.06743
80	3.149	3.982	-.11032
90	1.888	2.587	-.17486
95	1.061	1.580	-.23432
100	.090	0	-.62234

$$\text{L.E. radius} = 0.3966\left(\frac{\text{Thickness}}{\text{Chord}} \times \frac{1}{0.09}\right)^2$$

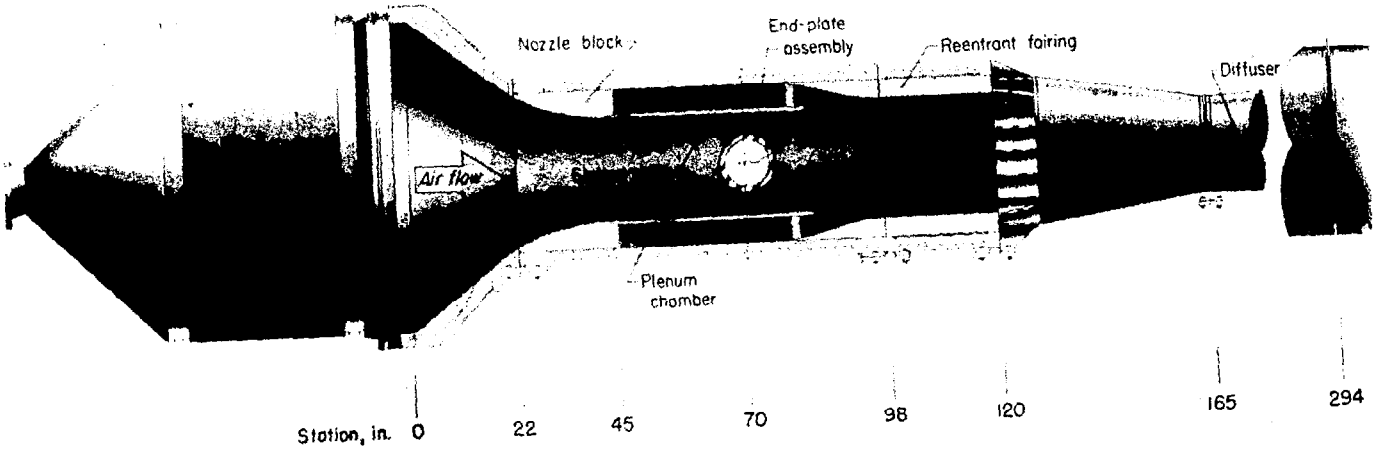
Slope of L.E. radius through end of chord
= 0.4212 × Design lift coefficient

[a]Values measured from and perpendicular to mean line; for other thicknesses multiply NACA 16-009 ordinates by $\left(\frac{\text{Thickness}}{\text{Chord}} \times \frac{1}{0.09}\right)$.

[b]For other design lift coefficients multiply mean-line characteristics by desired value of design lift coefficient.

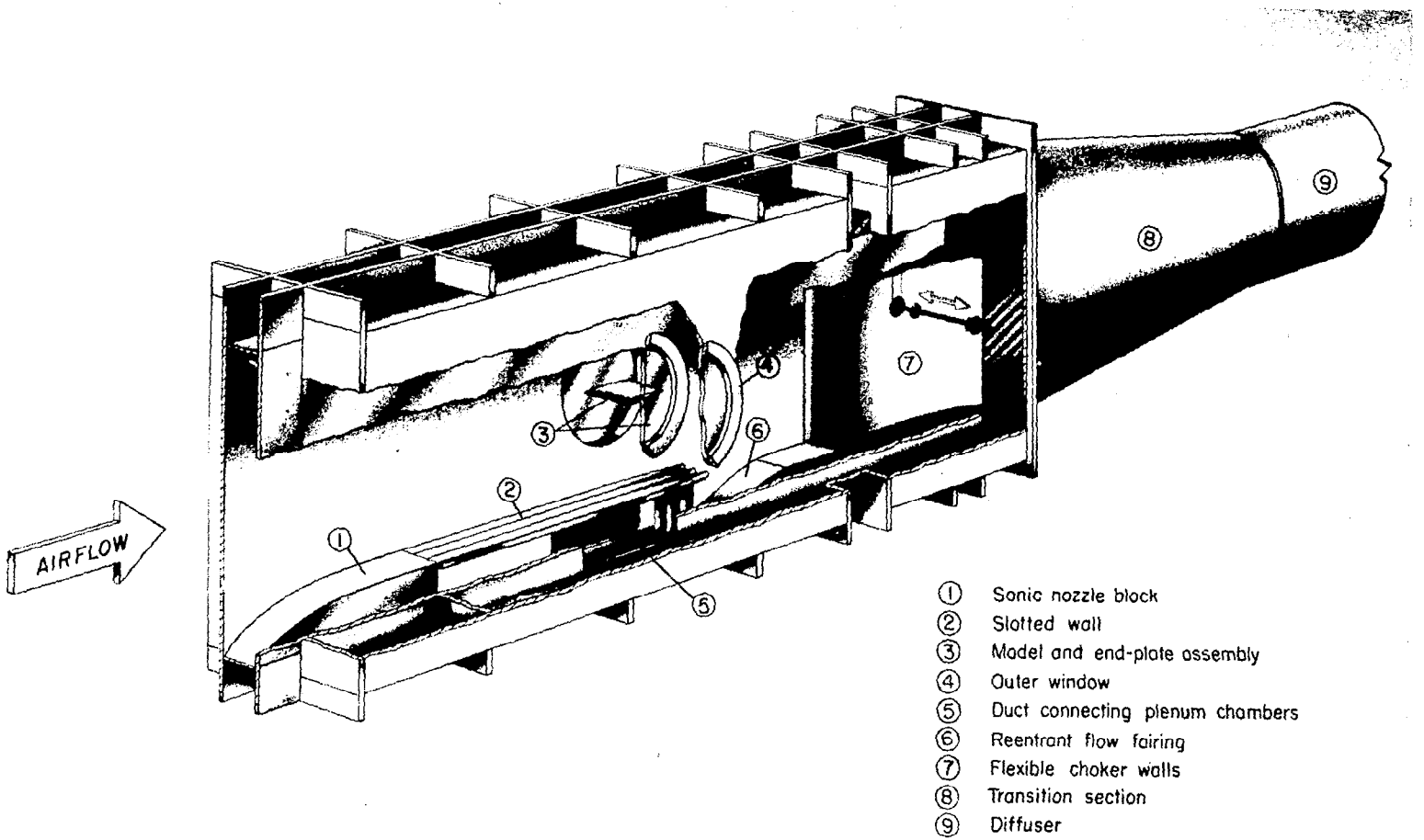
TABLE II.- INDEX TO PRESSURE-DISTRIBUTION DATA

NACA airfoil	Figure number for airfoil at angle of attack of -											
	-10°	-8°	-6°	-4°	-2°	0°	2°	4°	6°	8°	10°	12°
16-004	--	--	--	--	--	5	6	7	8	9	10	11
16-006	--	--	--	--	--	12	13	14	15	16	17	18
16-009	--	--	--	--	--	19	20	21	22	23	24	25
16-012	--	--	--	--	--	26	27	28	29	30	31	32
16-206	33	34	35	36	37	38	39	40	41	42	43	44
16-506	45	46	47	48	49	50	51	52	53	54	55	56
16-512	57	58	59	60	61	62	63	64	65	66	67	68



(a) General view. L-57-599

Figure 1.- Langley airfoil test apparatus.



(b) Test section. L-57-598

Figure 1.- Concluded.

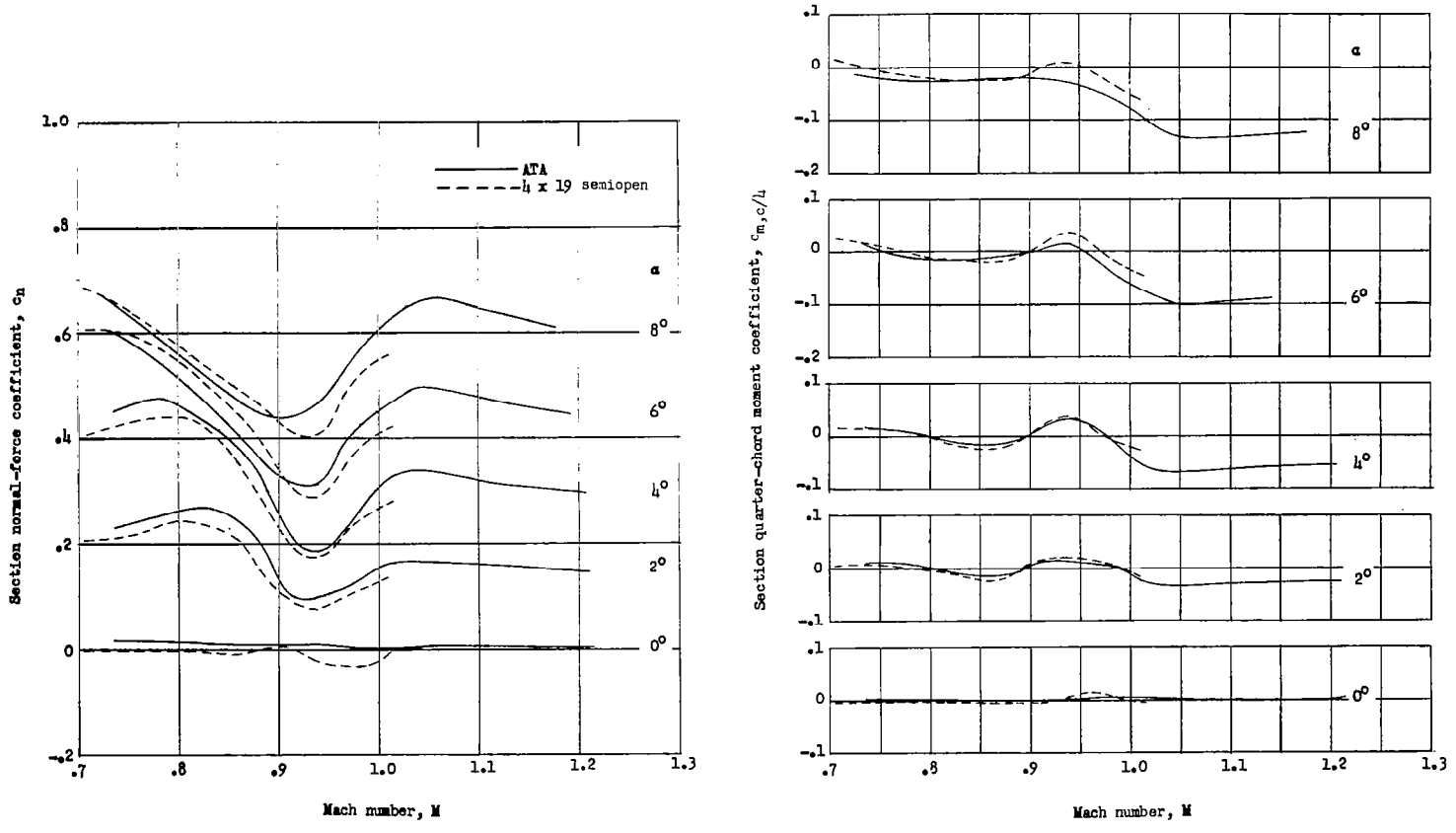
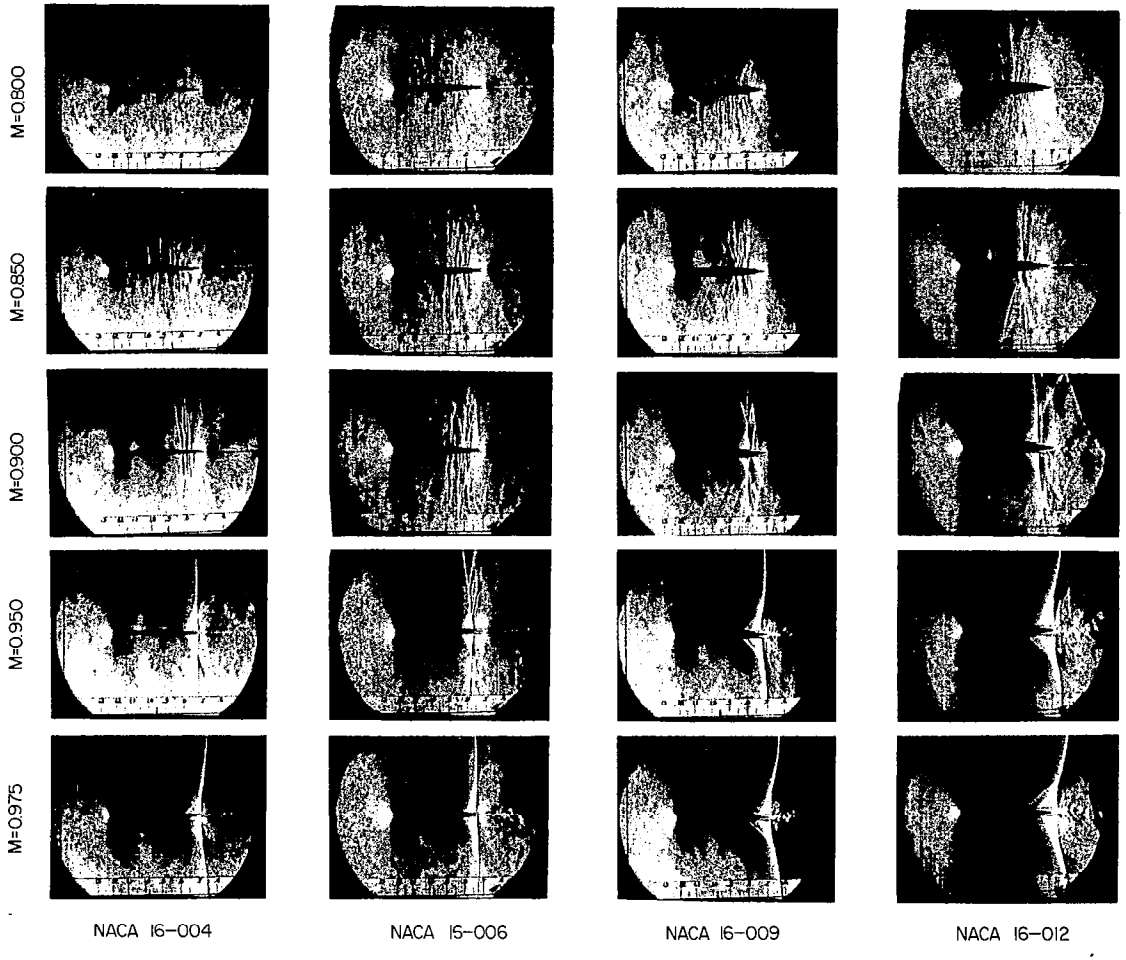
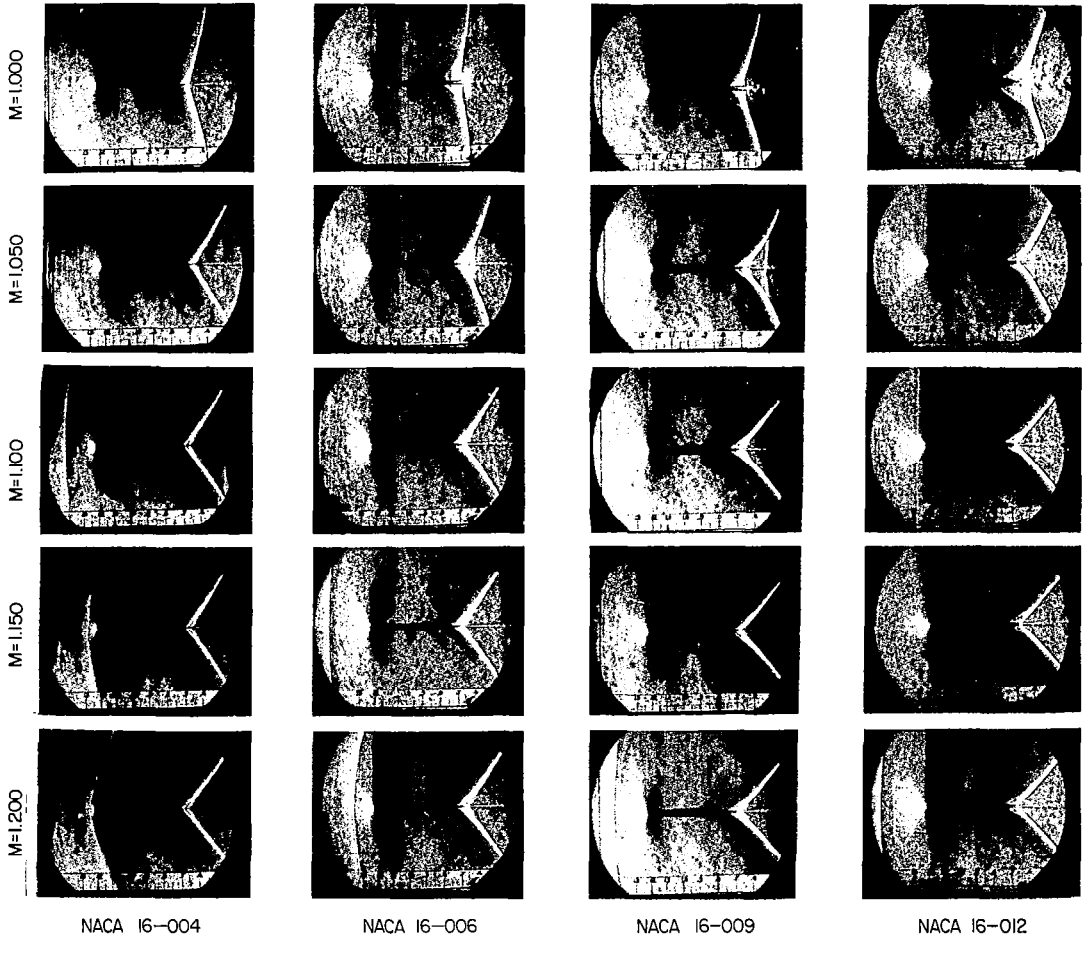


Figure 2.- Comparison of data obtained on an NACA 0012 airfoil in the ATA with data from the 4- by 19-inch semiopen tunnel.



(a) $\alpha = 0^{\circ}$. L-59-1869

Figure 3.- Schlieren flow photographs of symmetrical models.

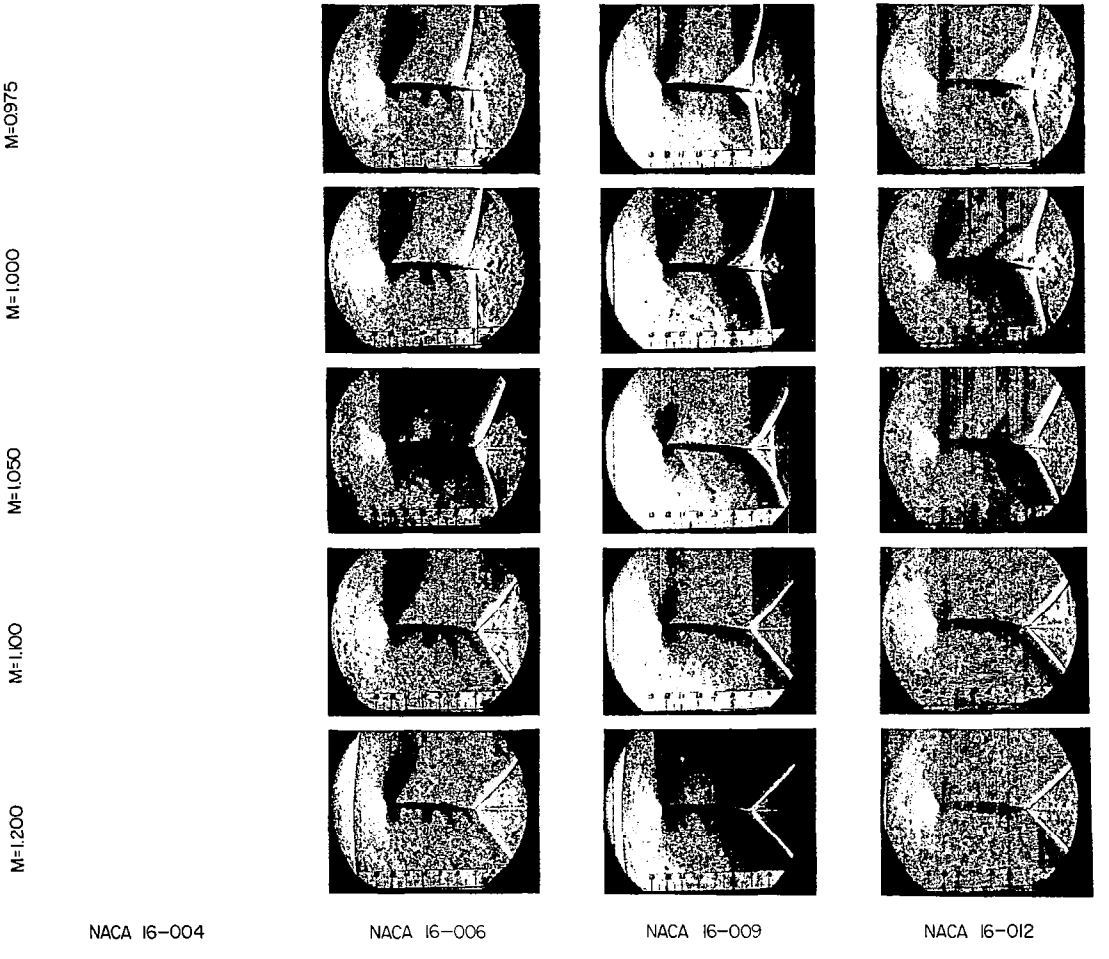


(a) $\alpha = 0^{\circ}$. Concluded. L-59-1870

Figure 3.- Continued.

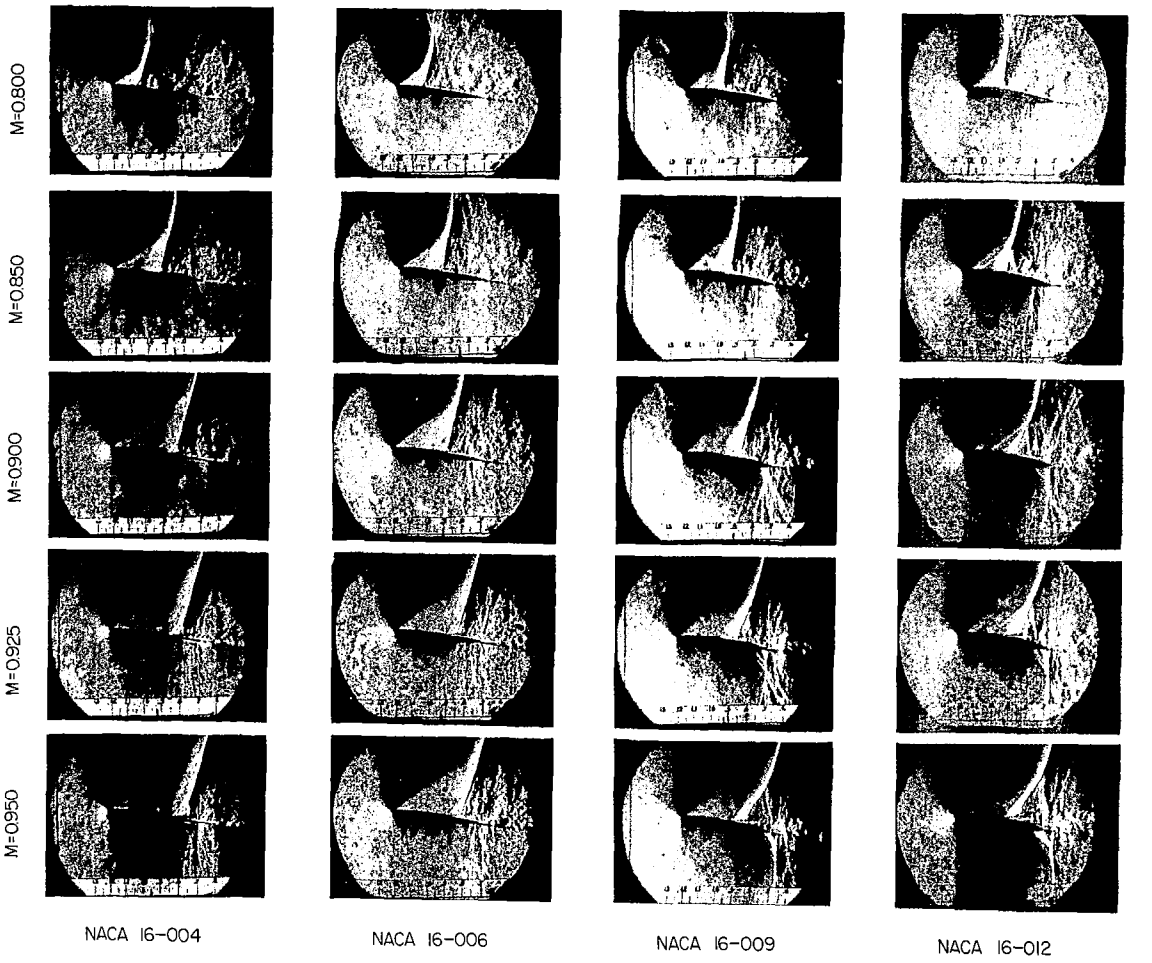
(b) $\alpha = 4^{\circ}$. L-59-1871

Figure 3.- Continued.



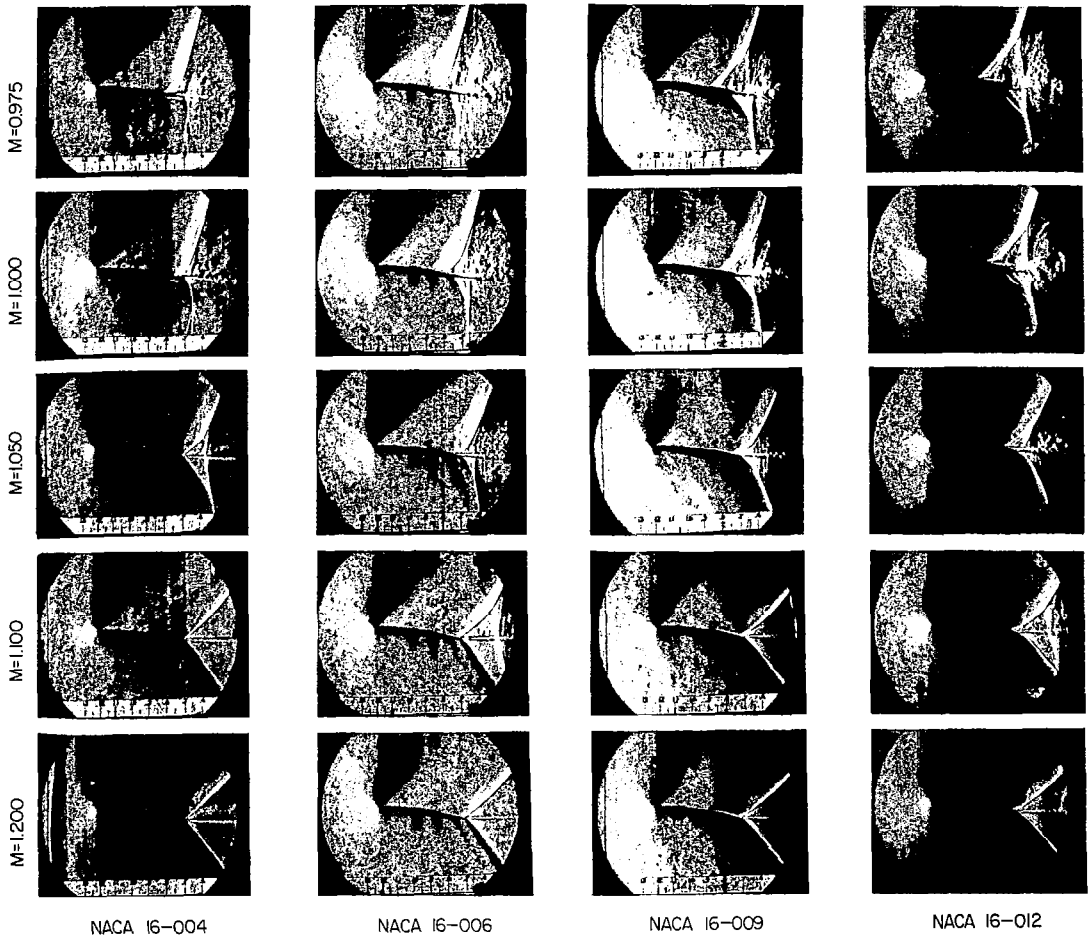
(b) $\alpha = 4^{\circ}$. Concluded. L-59-1872

Figure 3.- Continued.



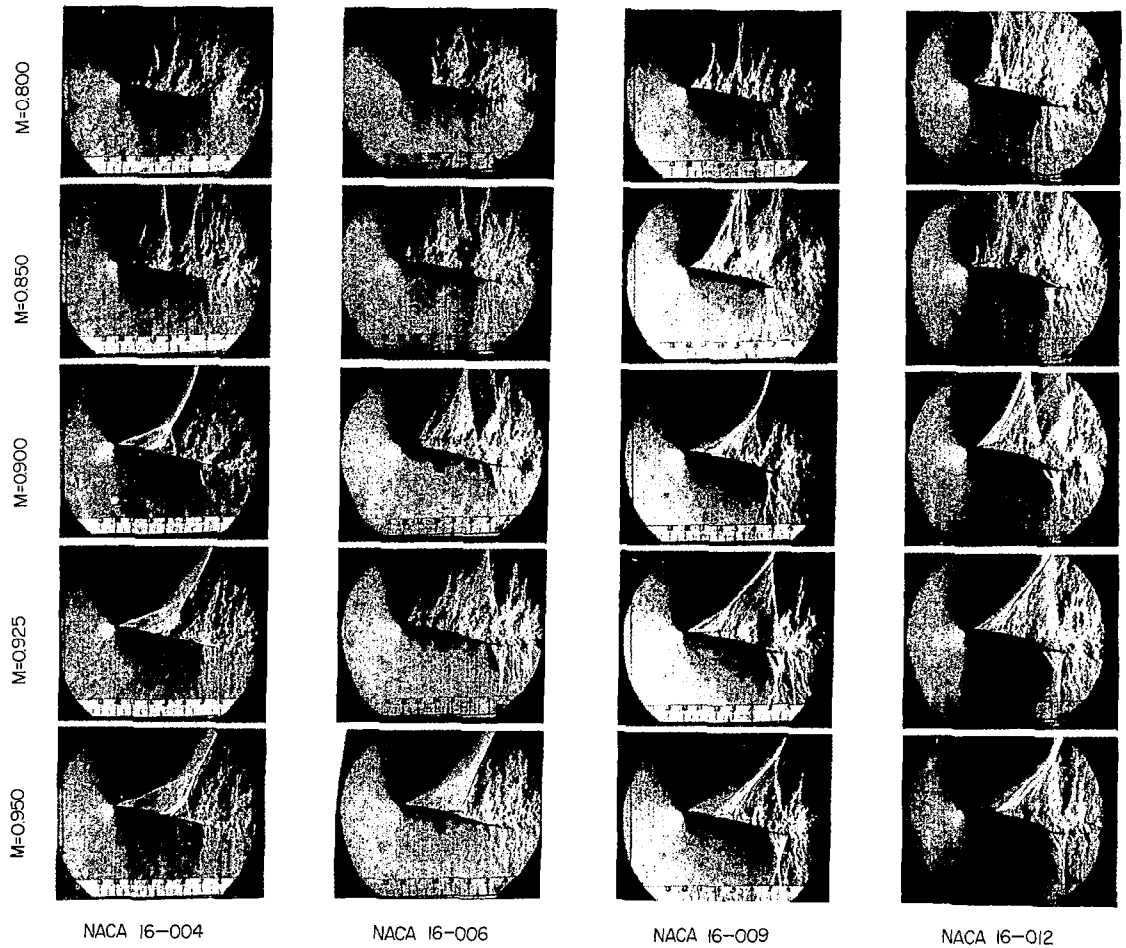
(c) $\alpha = 8^{\circ}$. L-59-1873

Figure 3.- Continued.



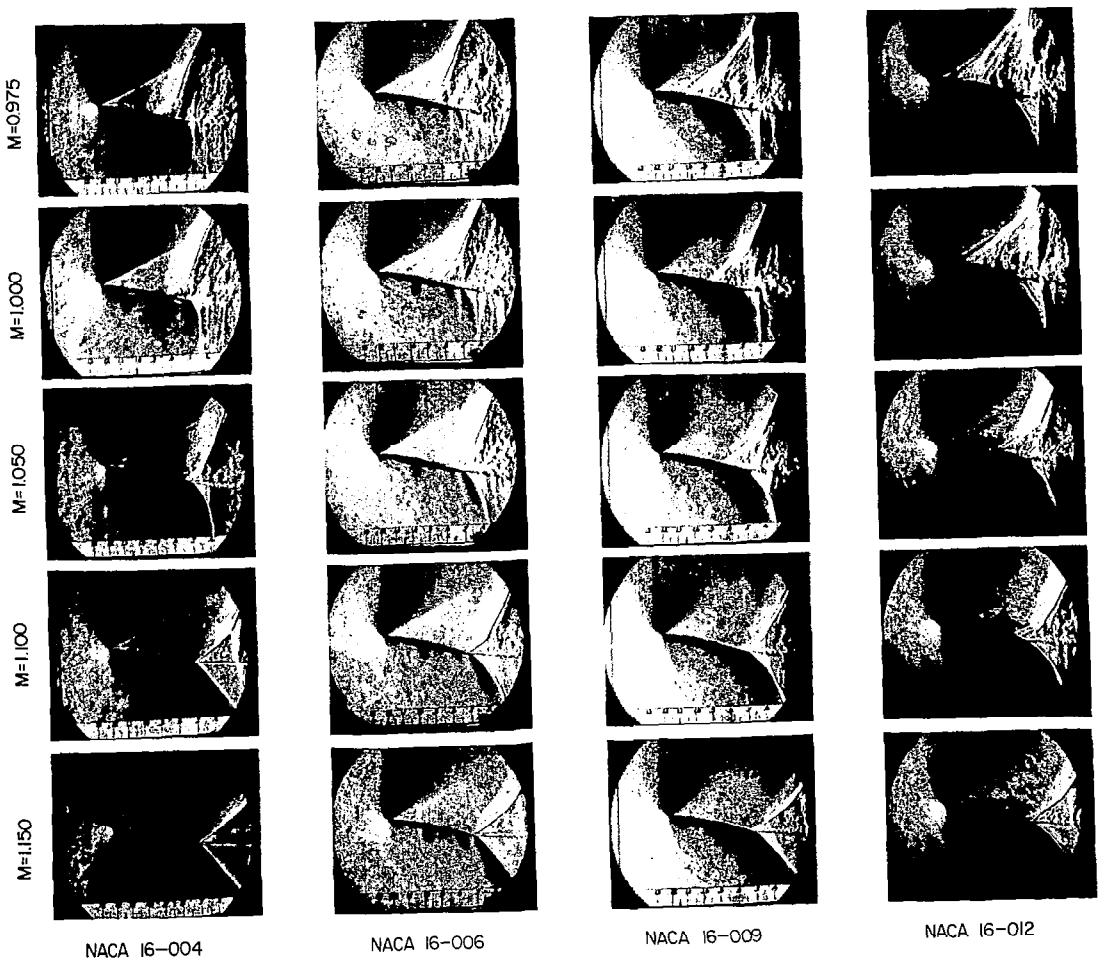
(c) $\alpha = 8^{\circ}$. Concluded. L-59-1874

Figure 3.- Continued.



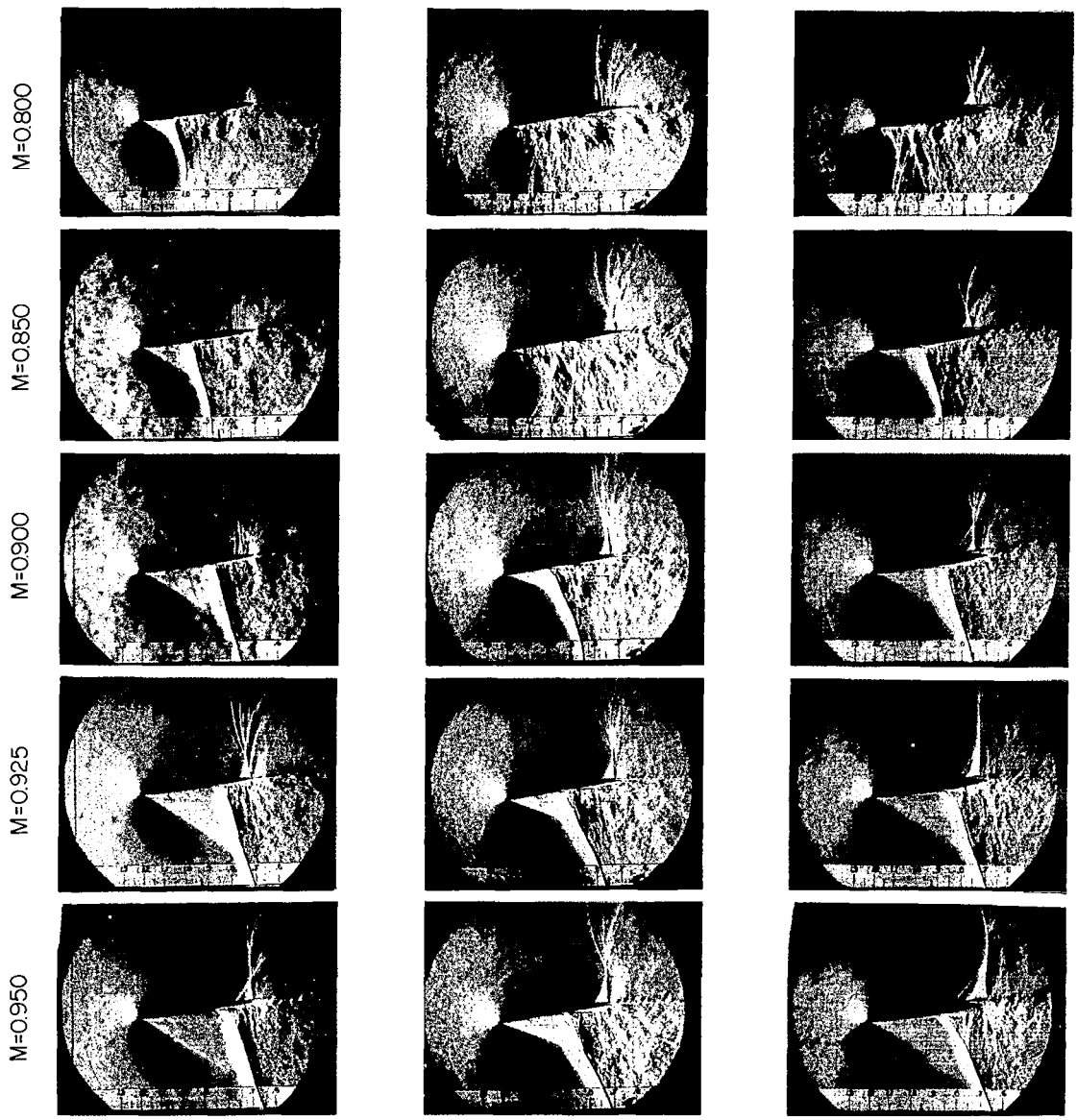
(d) $\alpha = 12^{\circ}$. L-59-1875

Figure 3.- Continued.



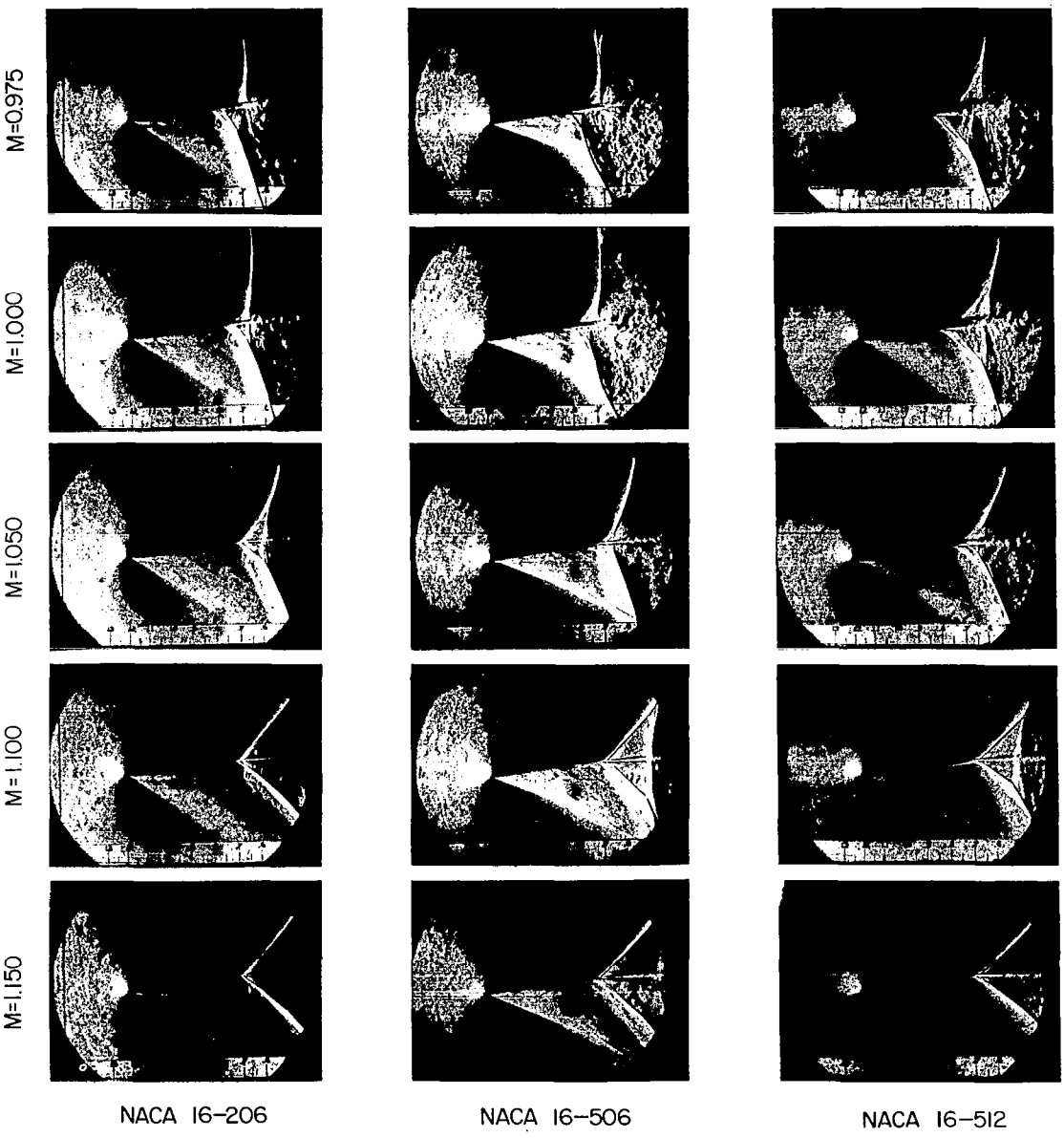
(d) $\alpha = 12^{\circ}$. Concluded. L-59-1876

Figure 3.- Concluded.



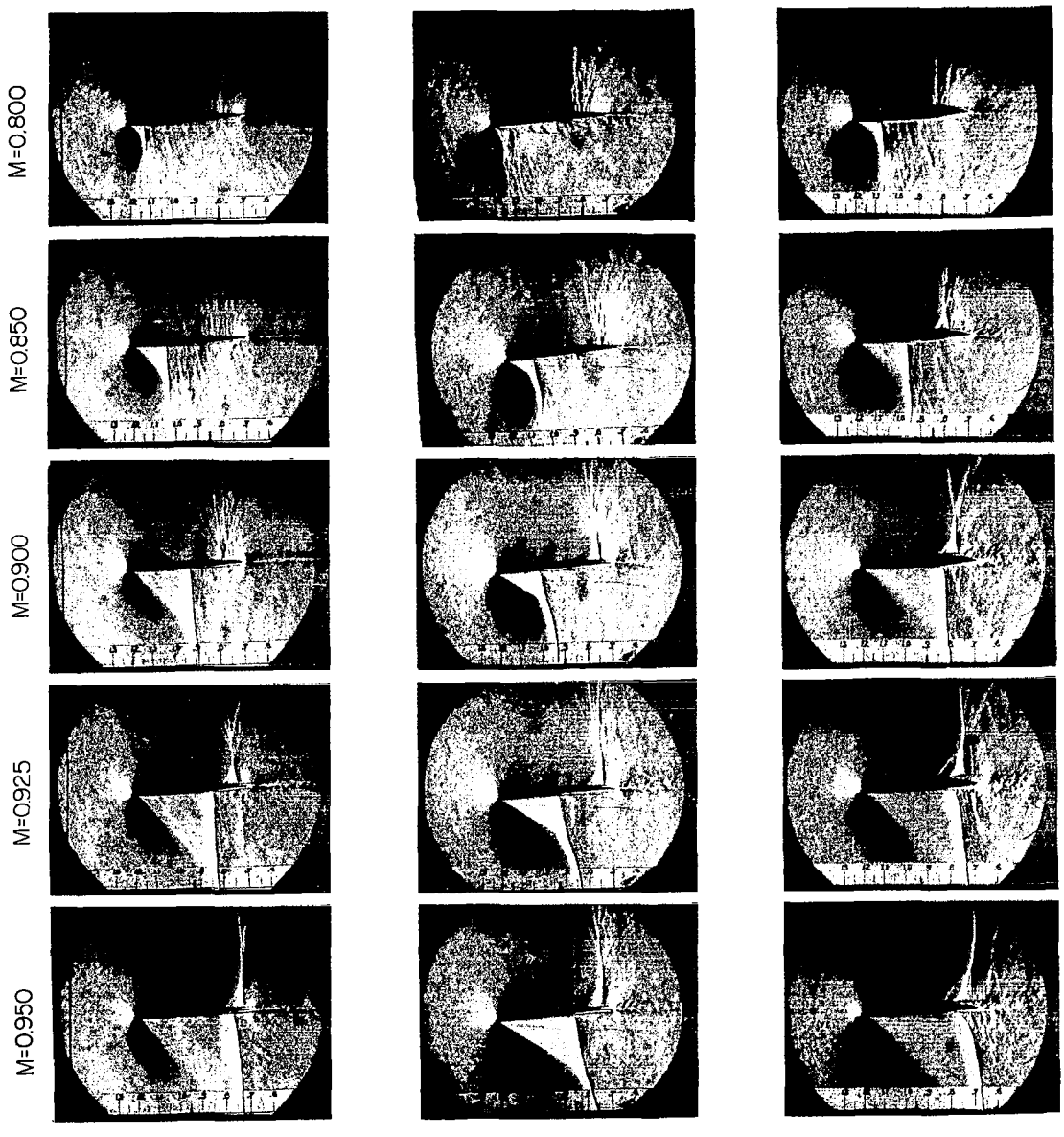
(a) $\alpha = -8^{\circ}$. L-59-1877

Figure 4.- Schlieren flow photographs of cambered models.



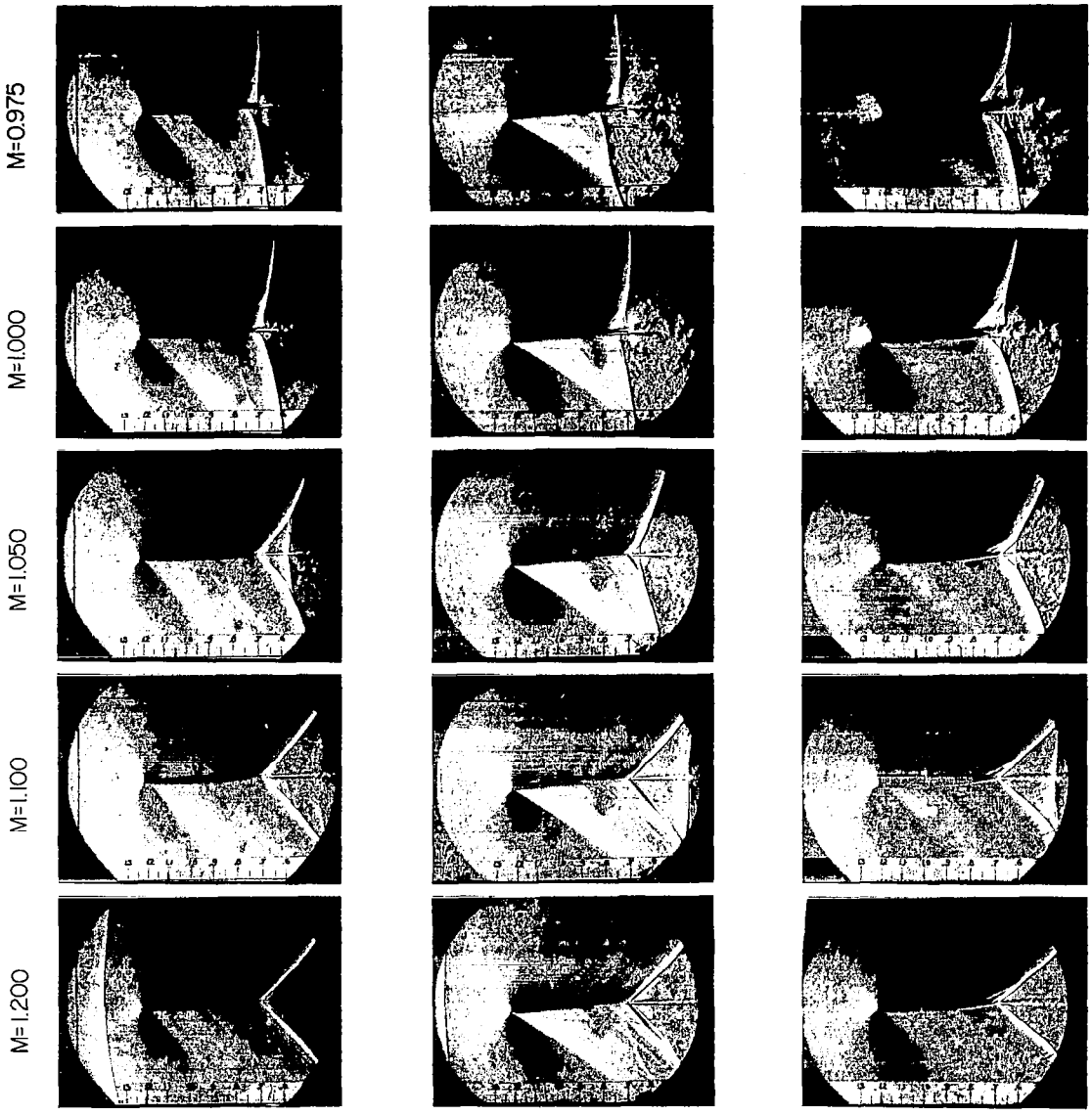
(a) $\alpha = -8^{\circ}$. Concluded. L-59-1878

Figure 4.- Continued.



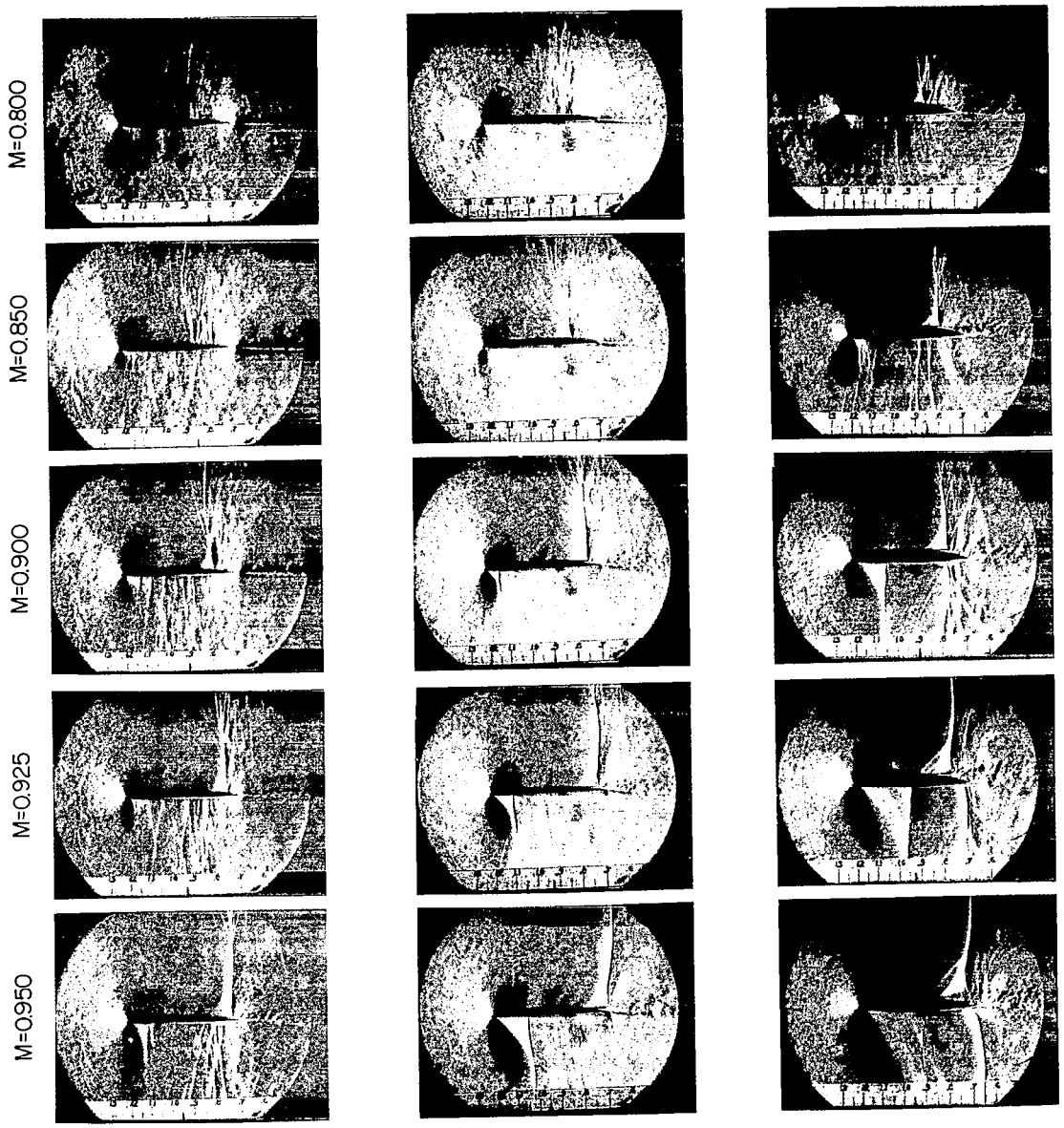
(b) $\alpha = -4^{\circ}$. L-59-1879

Figure 4.- Continued.



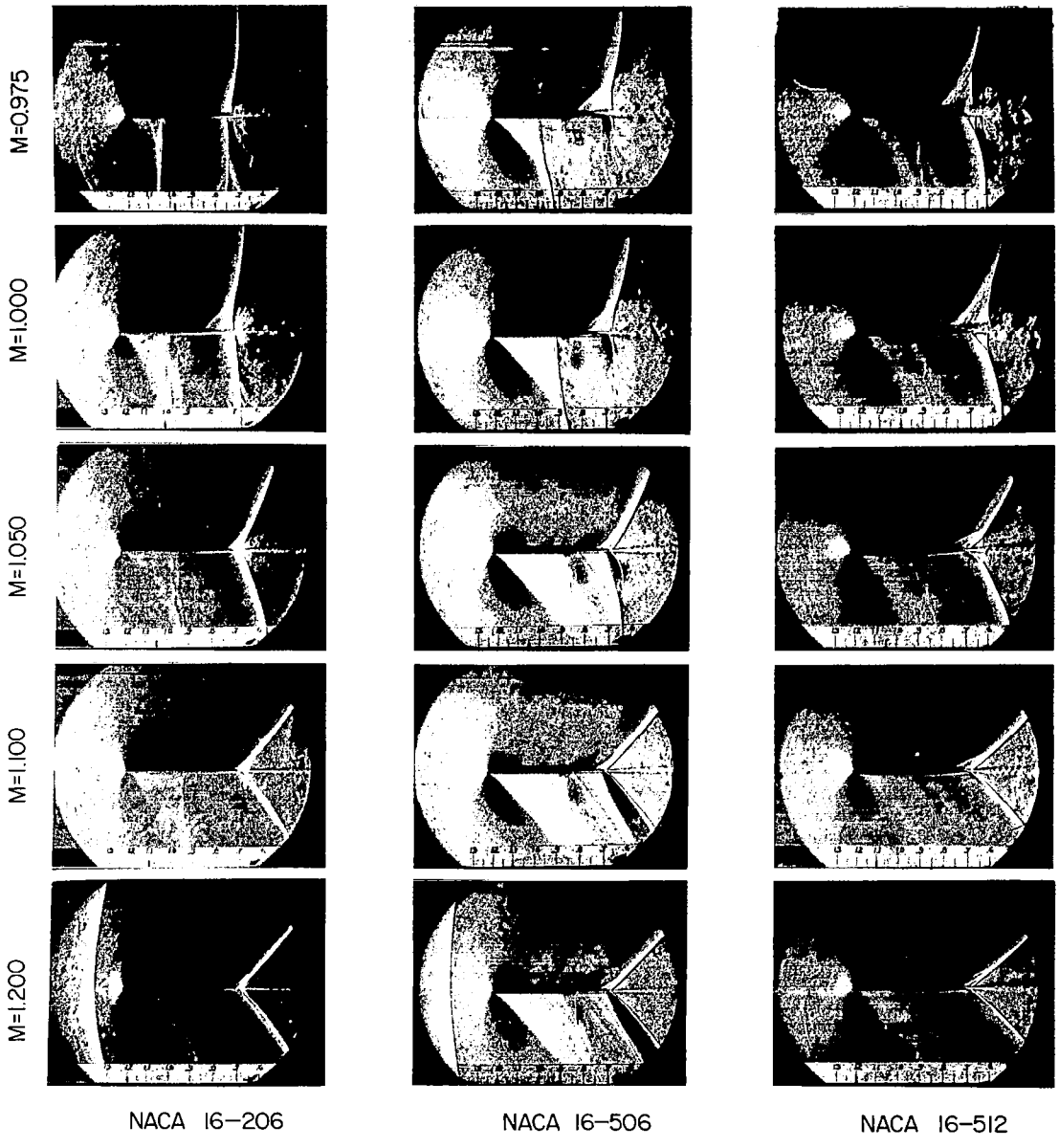
(b) $\alpha = -4^{\circ}$. Concluded. L-59-1880

Figure 4.- Continued.



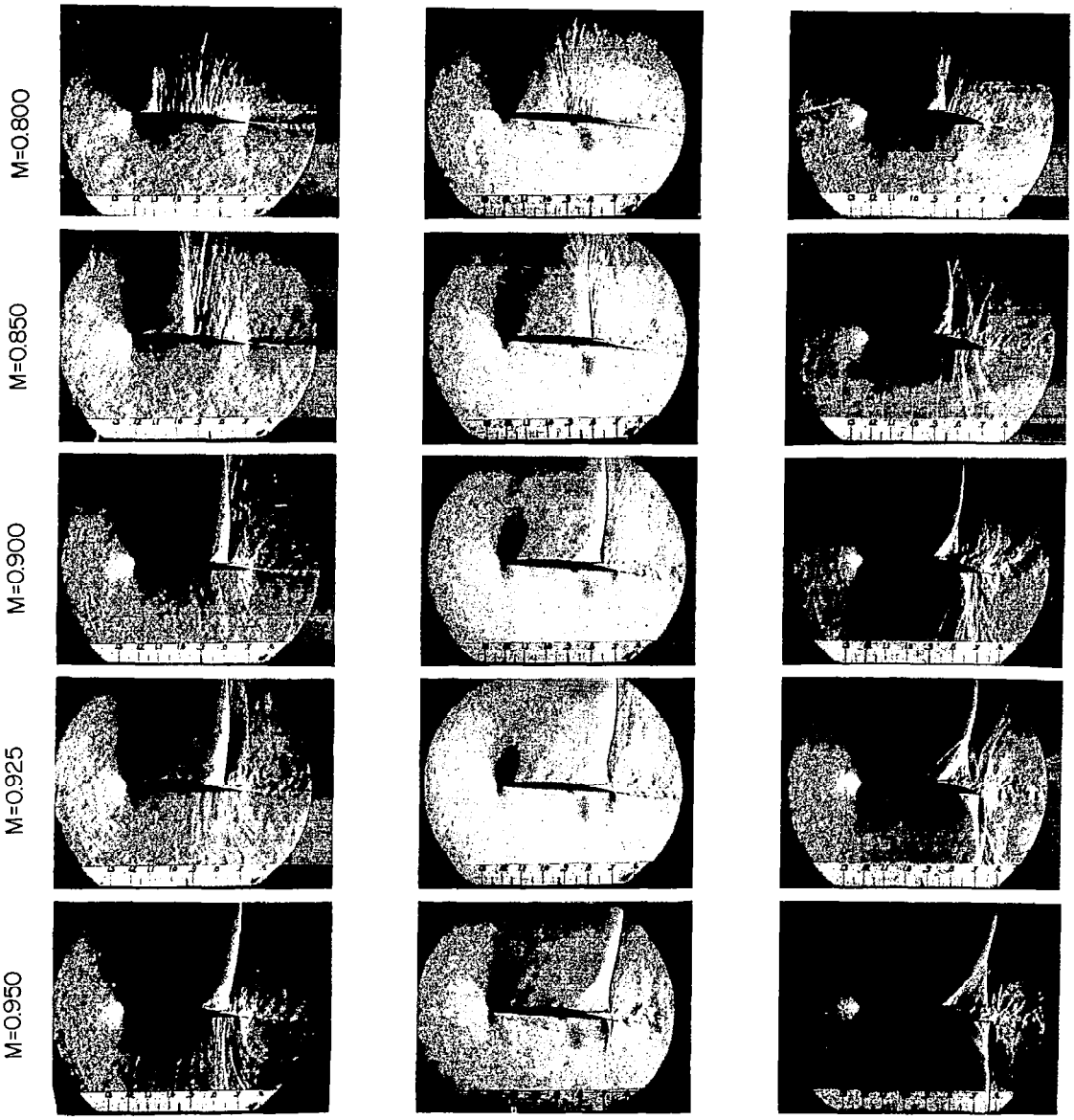
(c) $\alpha = 0^{\circ}$. L-59-1881

Figure 4.- Continued.



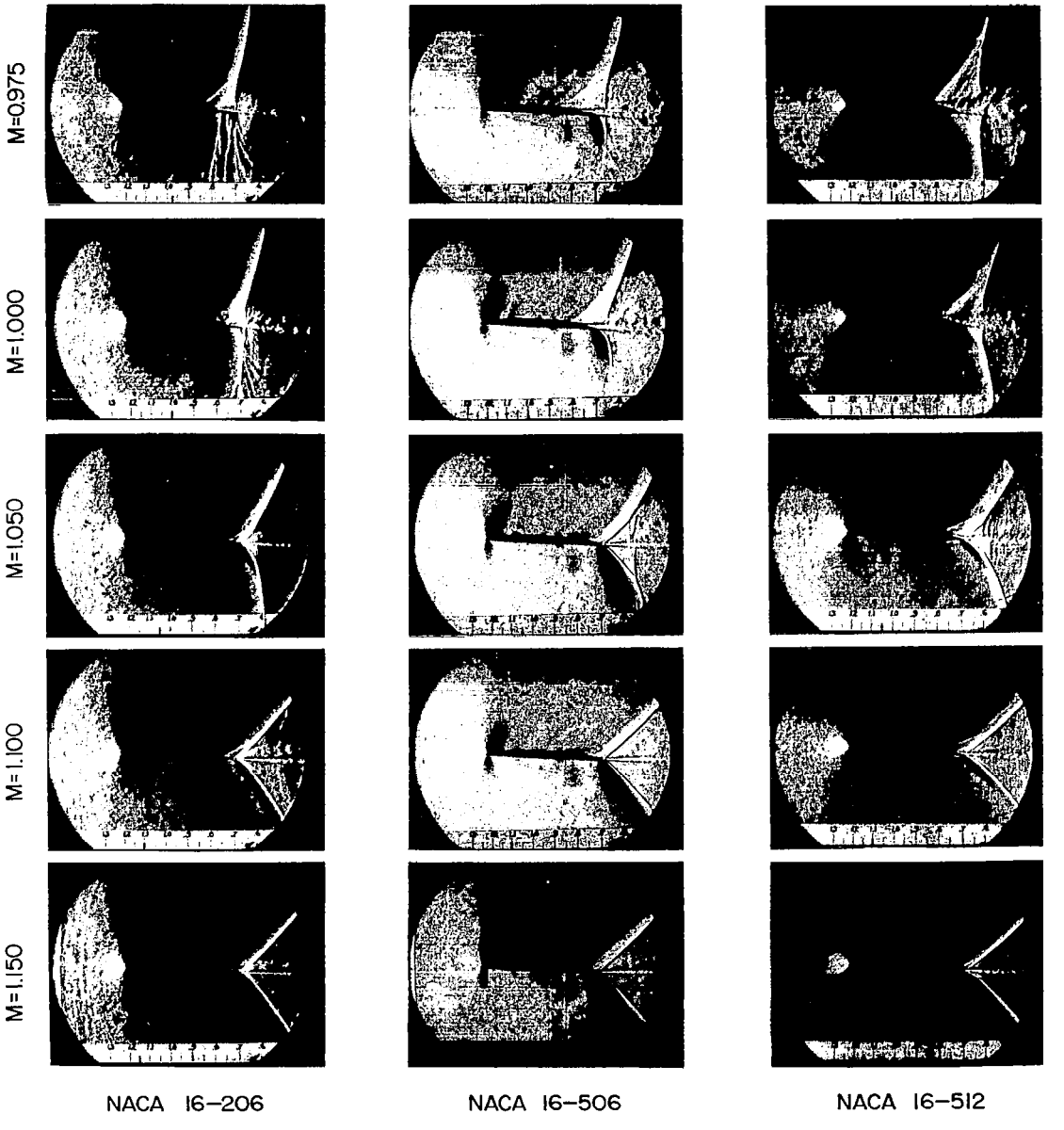
(c) $\alpha = 0^{\circ}$. Concluded. L-59-1882

Figure 4.- Continued.



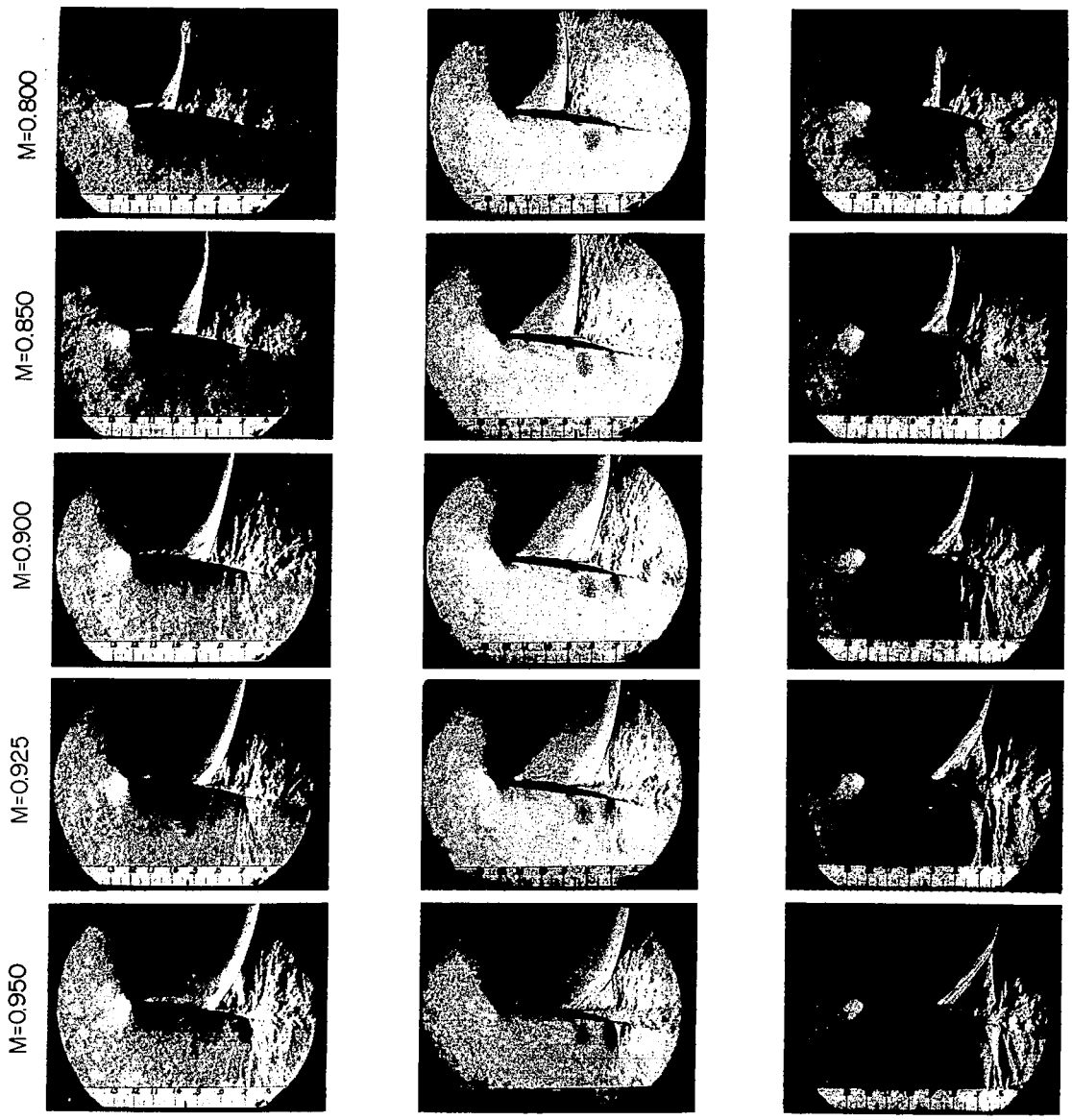
(d) $\alpha = 4^{\circ}$. L-59-1883

Figure 4.- Continued.



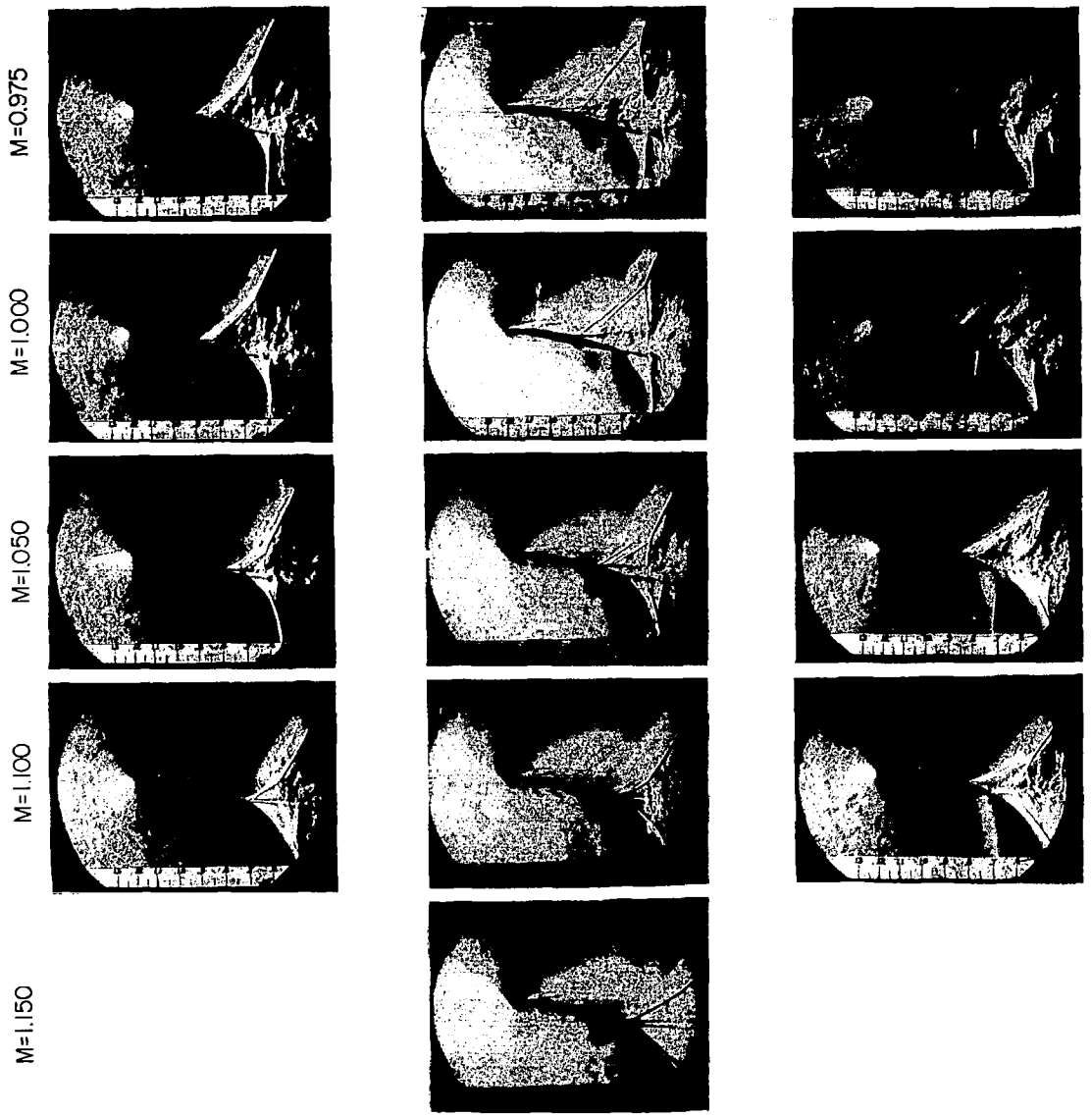
(d) $\alpha = 4^{\circ}$. Concluded. L-59-1884

Figure 4.- Continued.



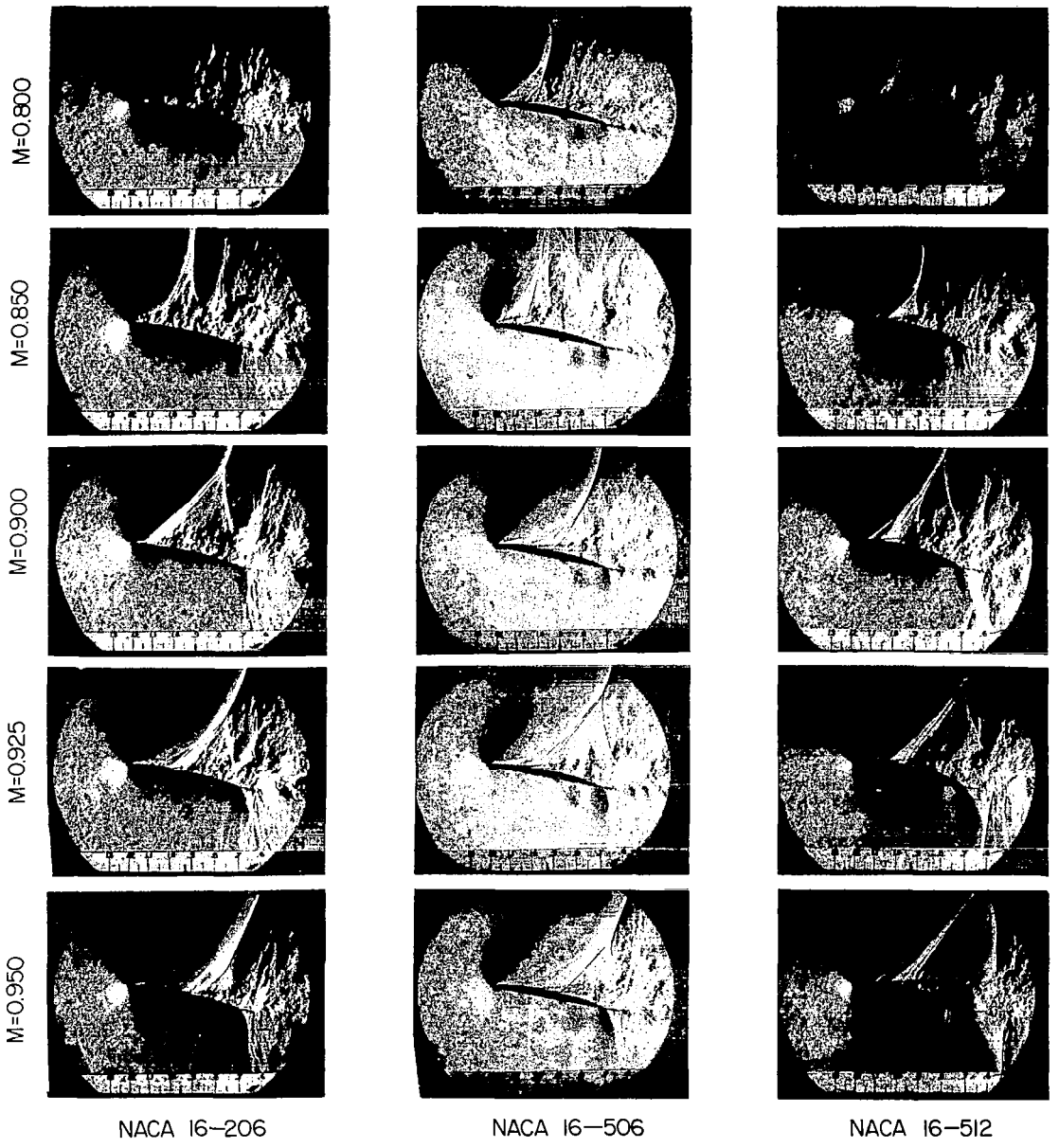
(e) $\alpha = 8^{\circ}$. L-59-1885

Figure 4.- Continued.



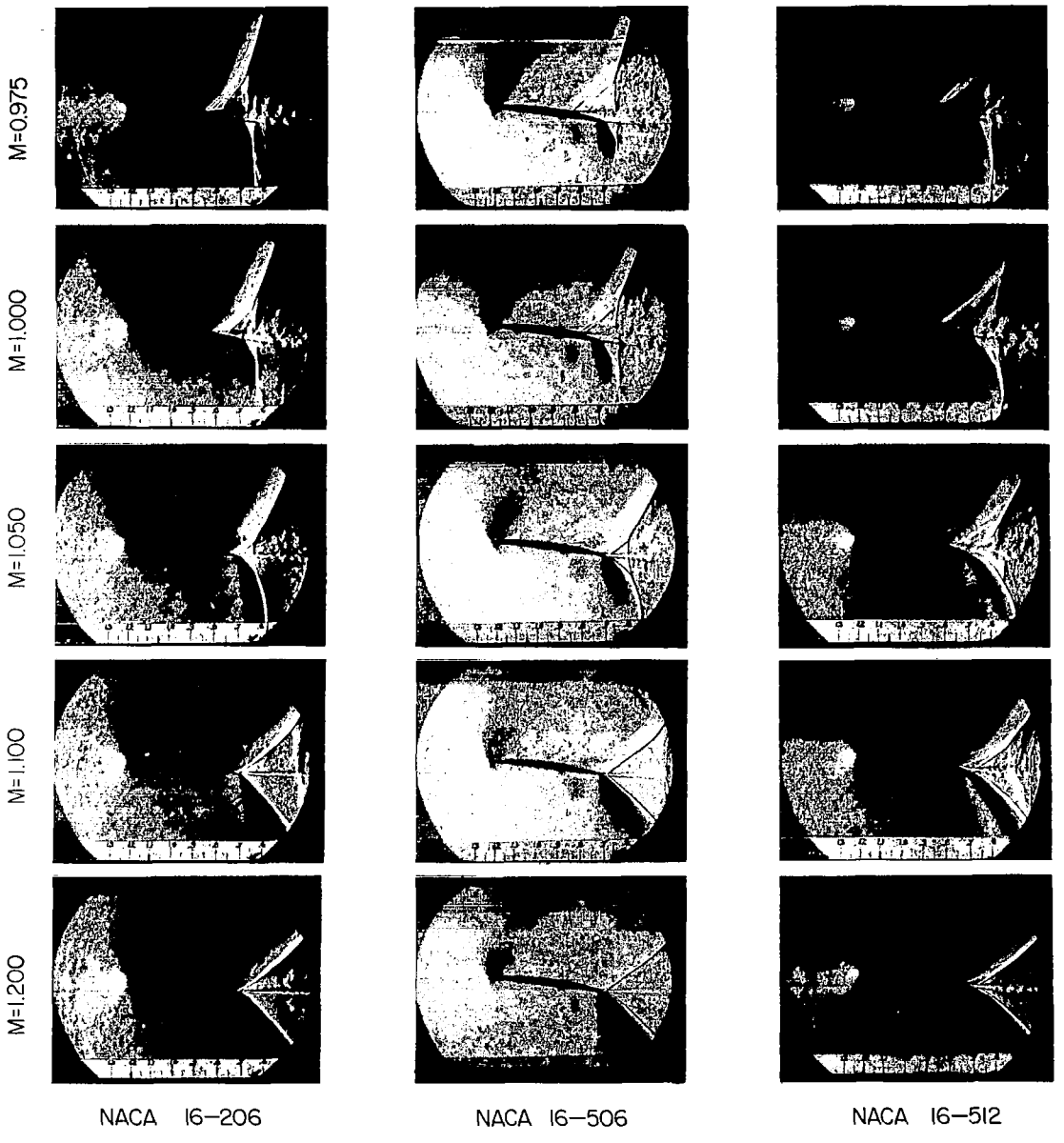
(e) $\alpha = 8^{\circ}$. Concluded. L-59-1886

Figure 4.- Continued.



(f) $\alpha = 12^{\circ}$. L-59-1887

Figure 4.- Continued.



(f) $\alpha = 12^{\circ}$. Concluded. L-59-1888

Figure 4.- Concluded.

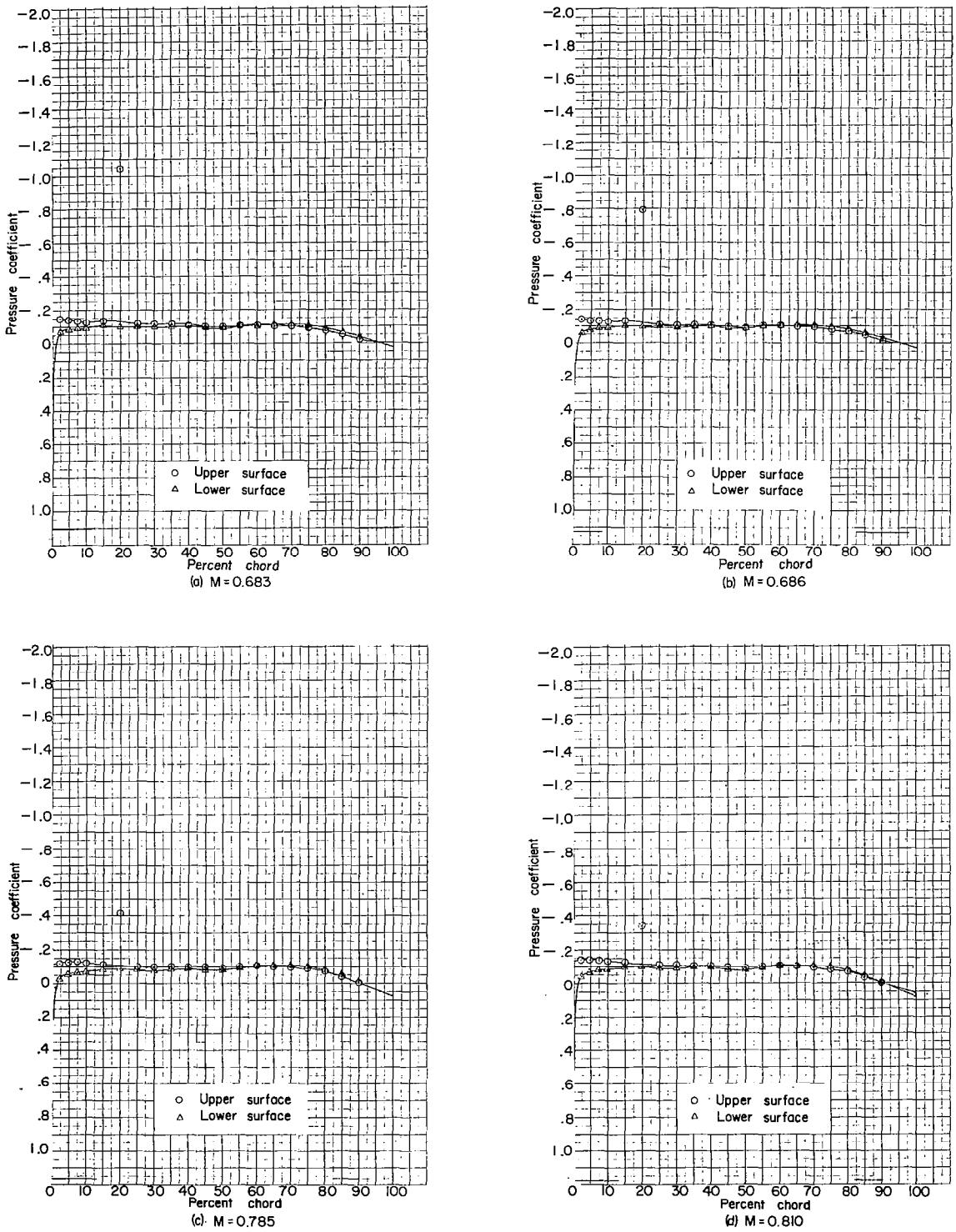
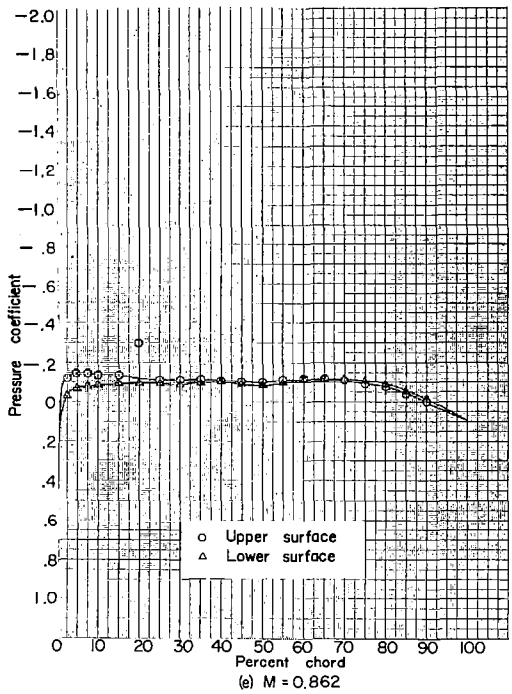
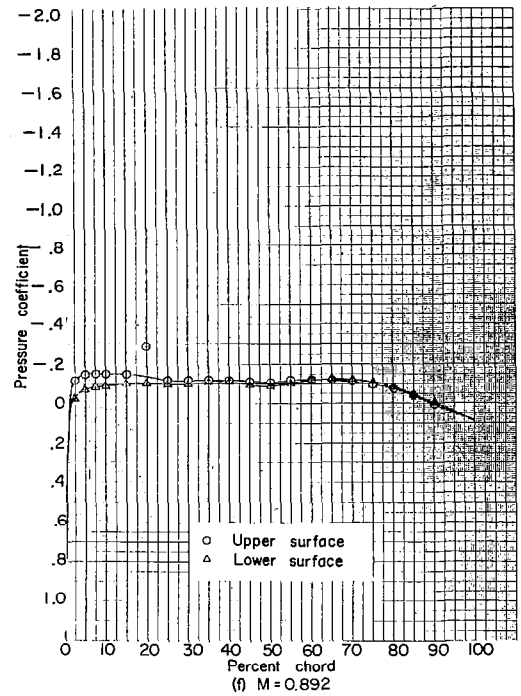
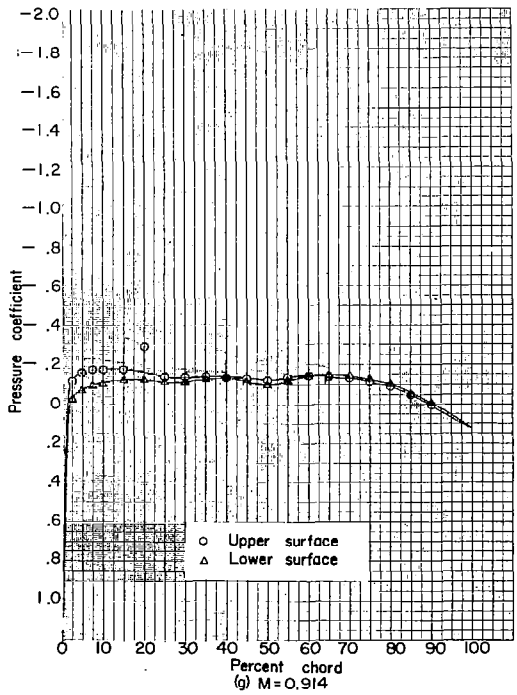


Figure 5.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 0^{\circ}$.







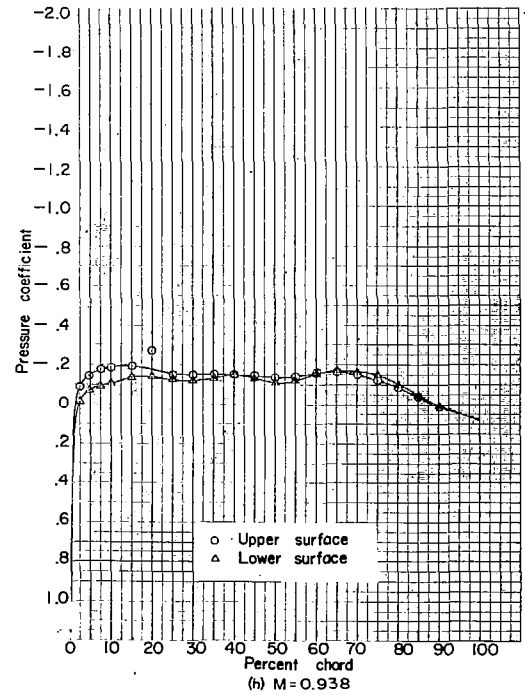
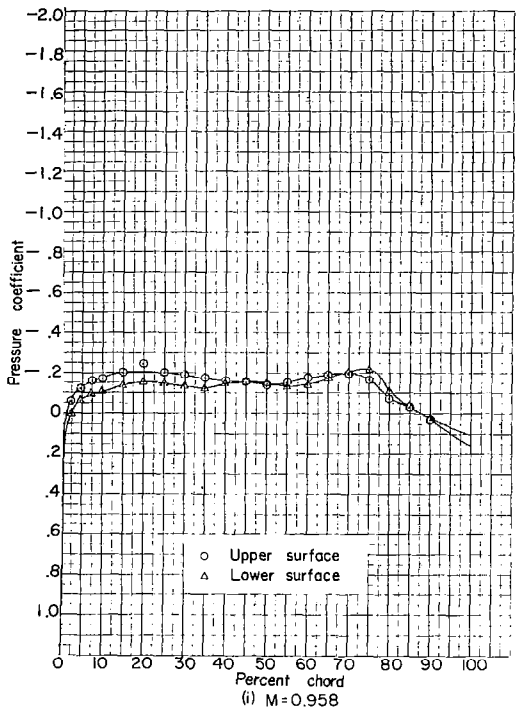
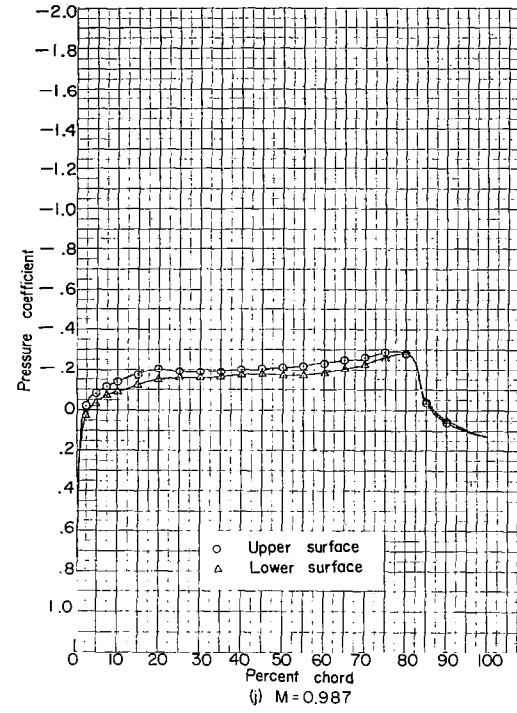
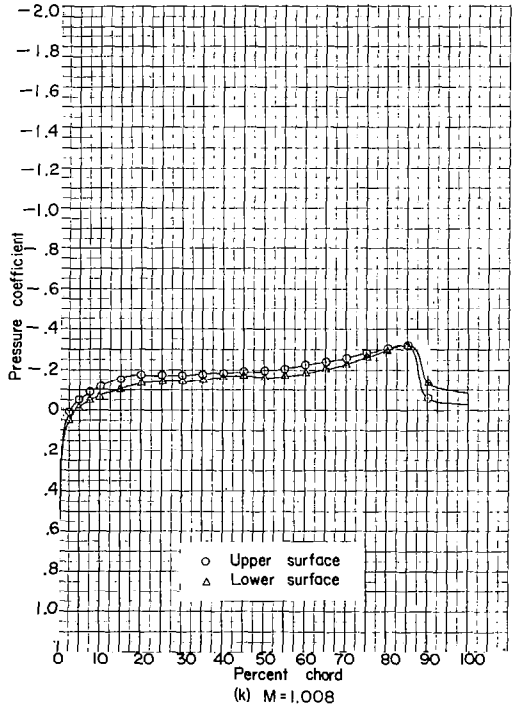


Figure 5.- Continued. NACA 16-004; $\alpha = 0^{\circ}$.







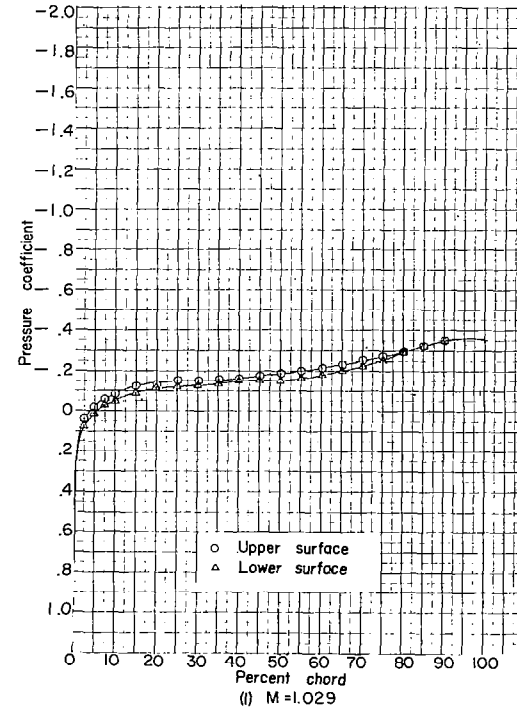
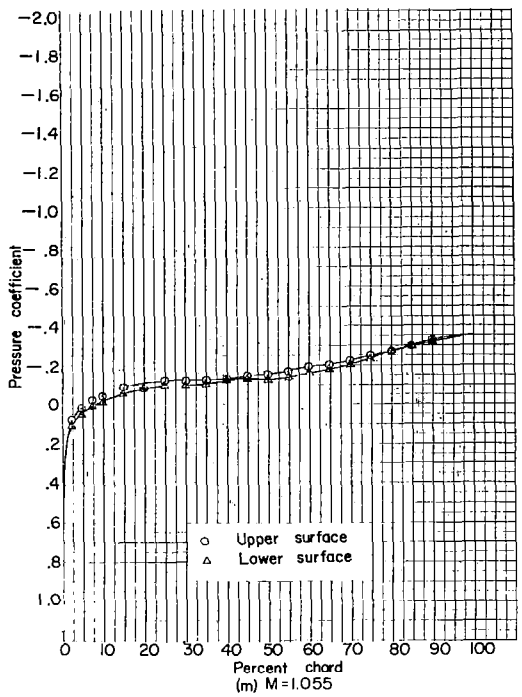
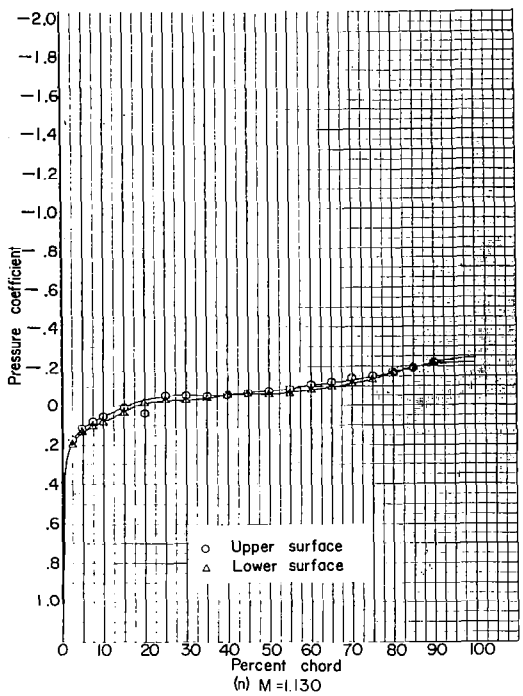
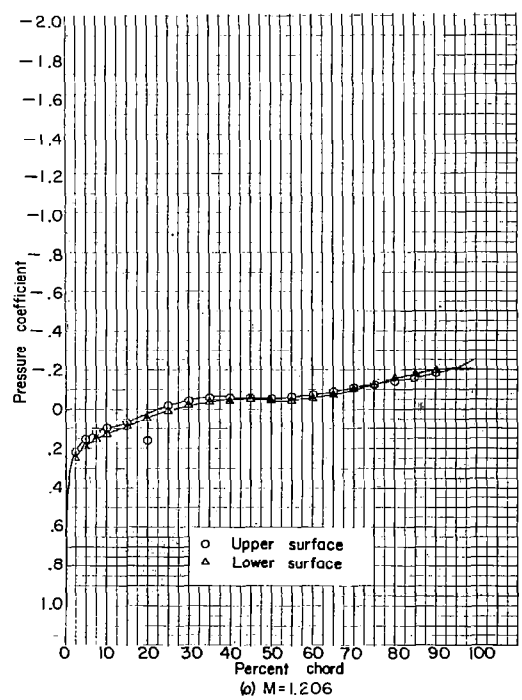


Figure 5.- Continued. NACA 16-004; $\alpha = 0^{\circ}$.







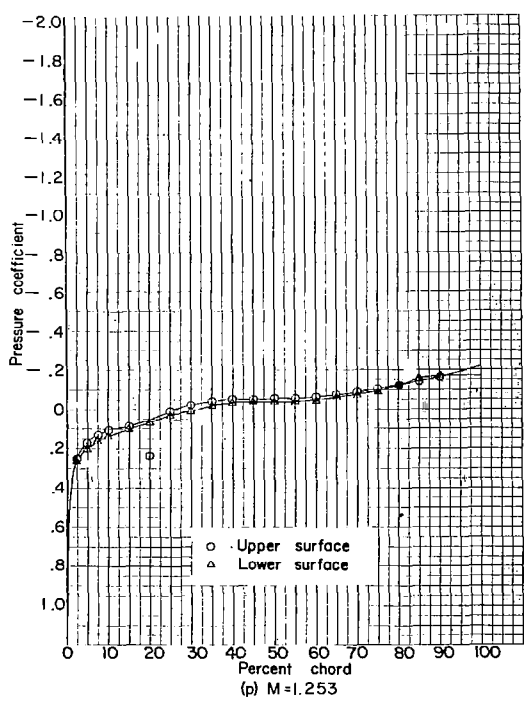


Figure 5.- Concluded. NACA 16-004; $\alpha = 0^{\circ}$.

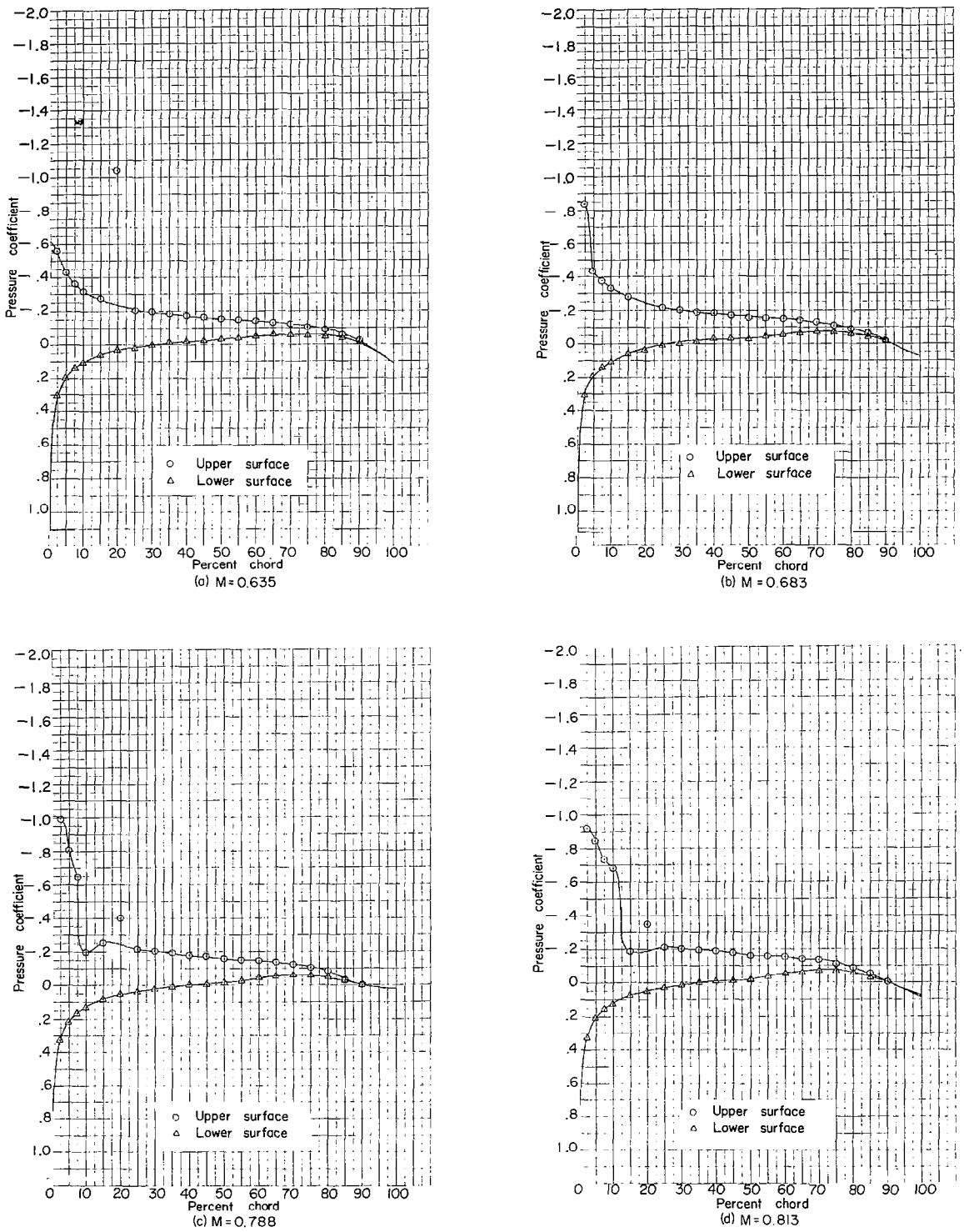
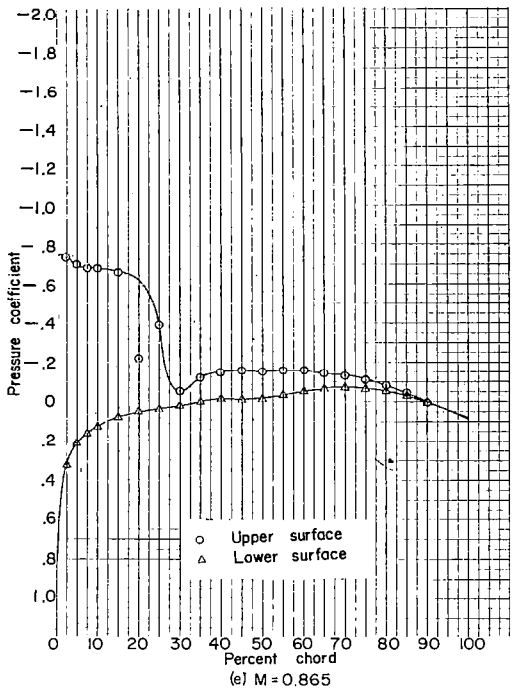
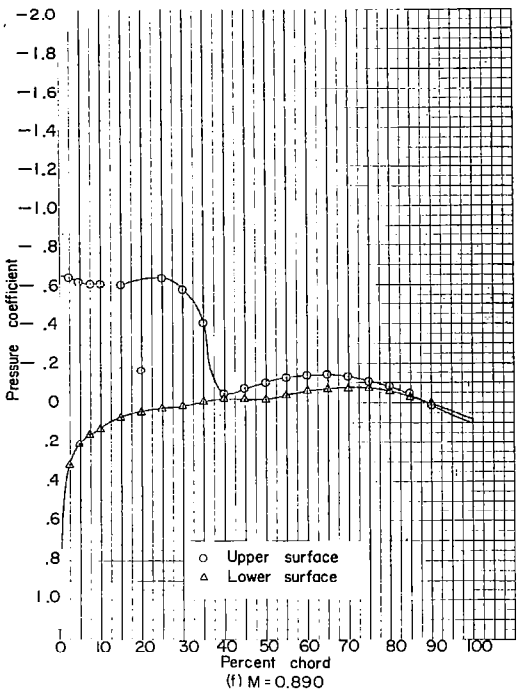
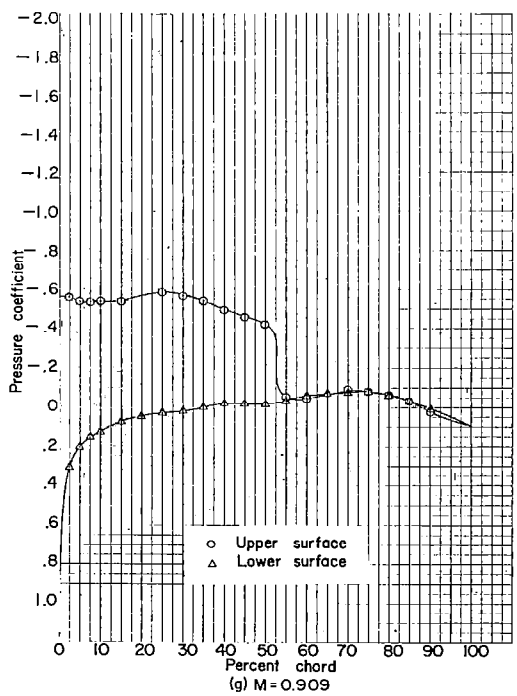


Figure 6.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 2^{\circ}$.







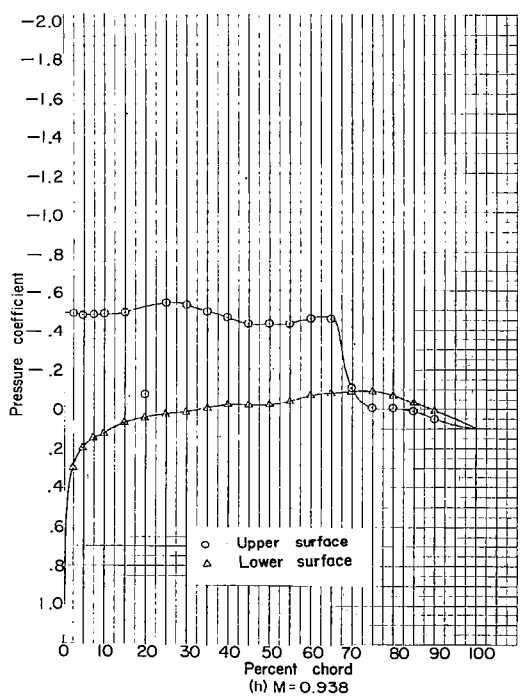
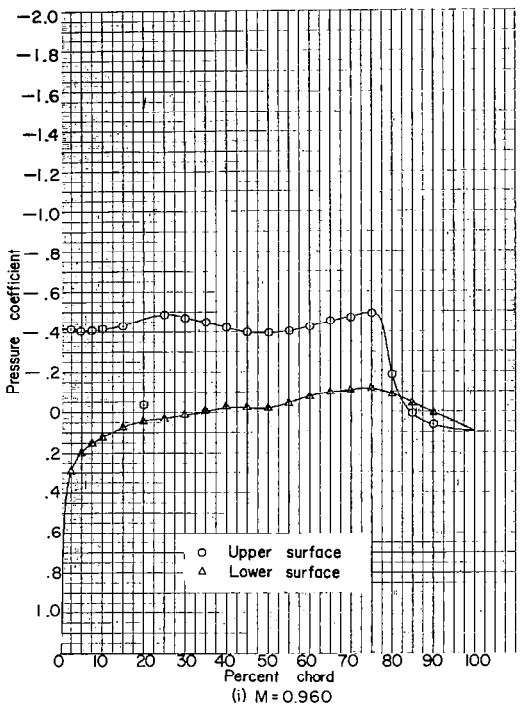
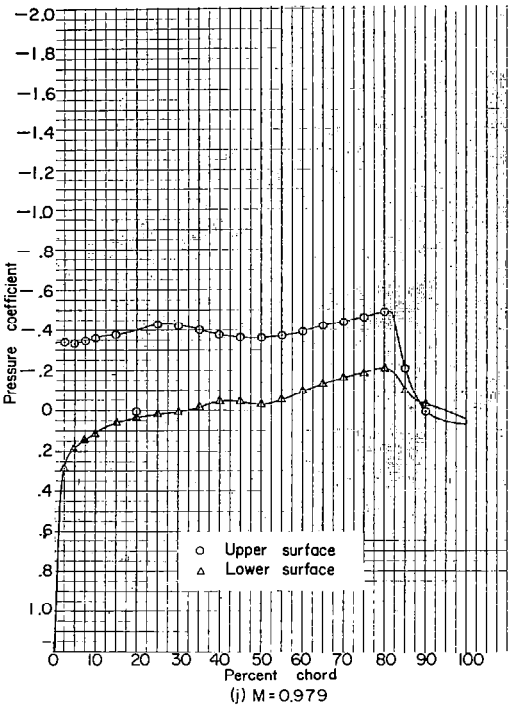


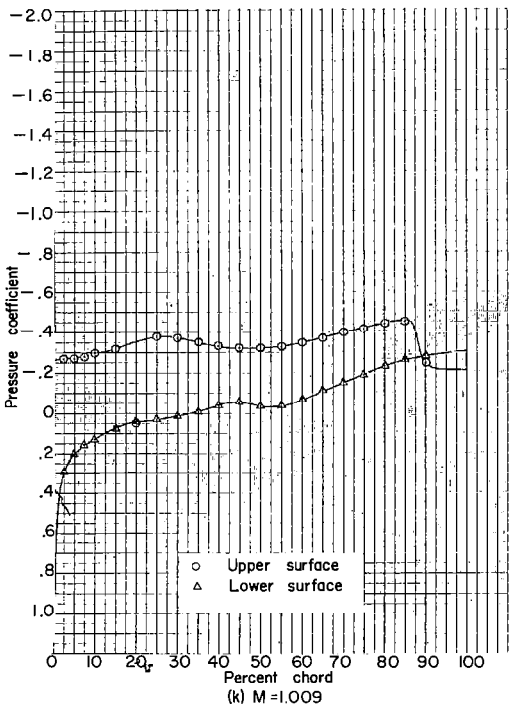
Figure 6.- Continued. NACA 16-004; $\alpha = 2^{\circ}$.



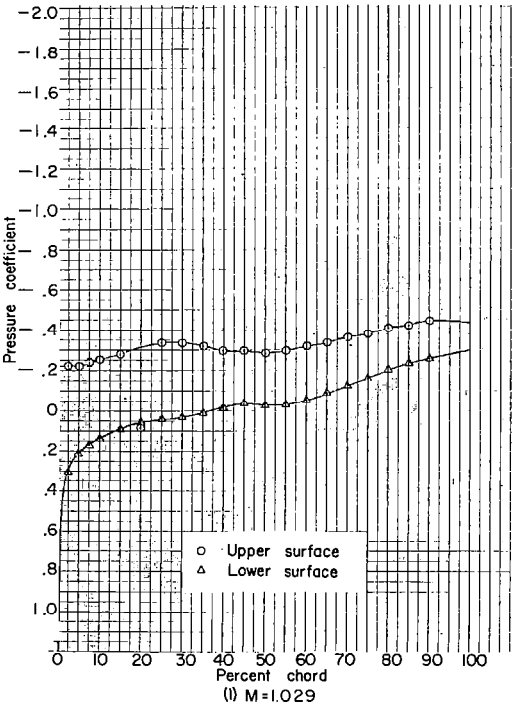
(i) M = 0.960



(j) M = 0.979

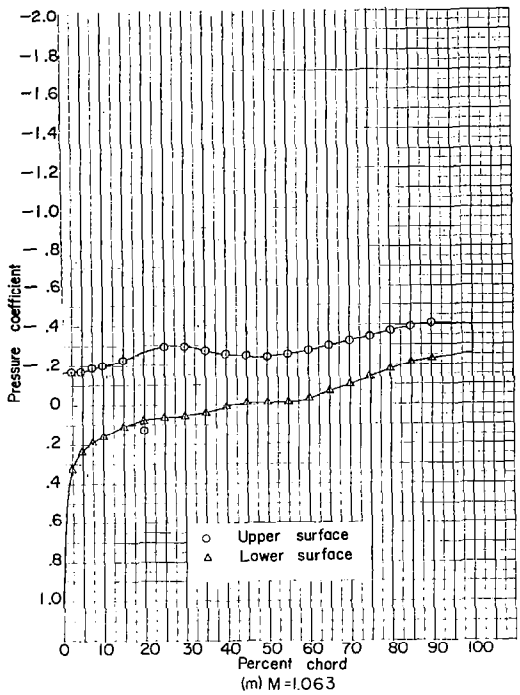


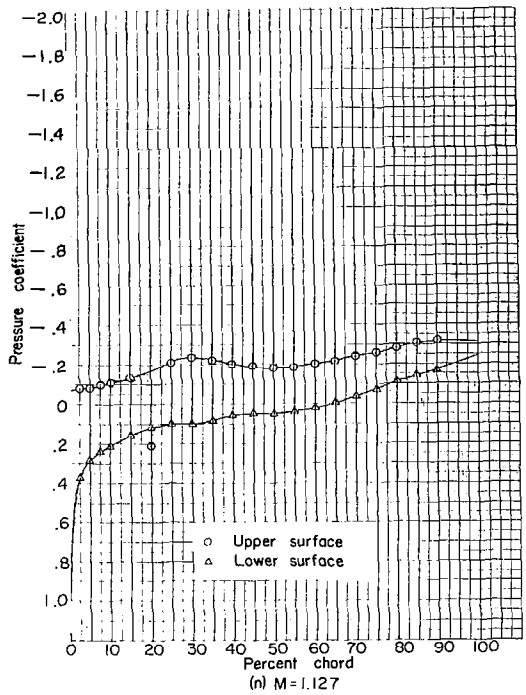
(k) M = 1.009

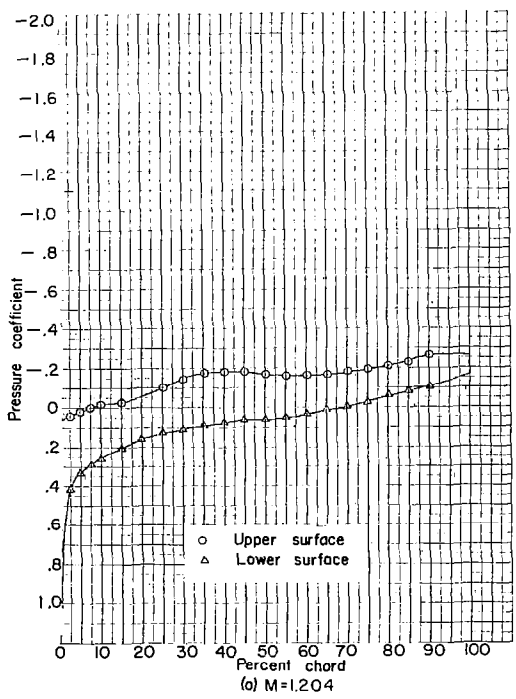


(l) M = 1.029

Figure 6.- Continued. NACA 16-004; $\alpha = 2^{\circ}$.







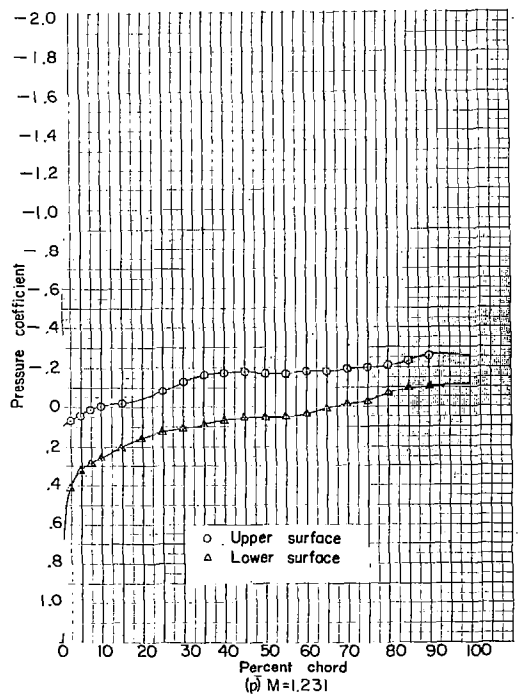


Figure 6.- Concluded. NACA 16-004; $\alpha = 2^{\circ}$.

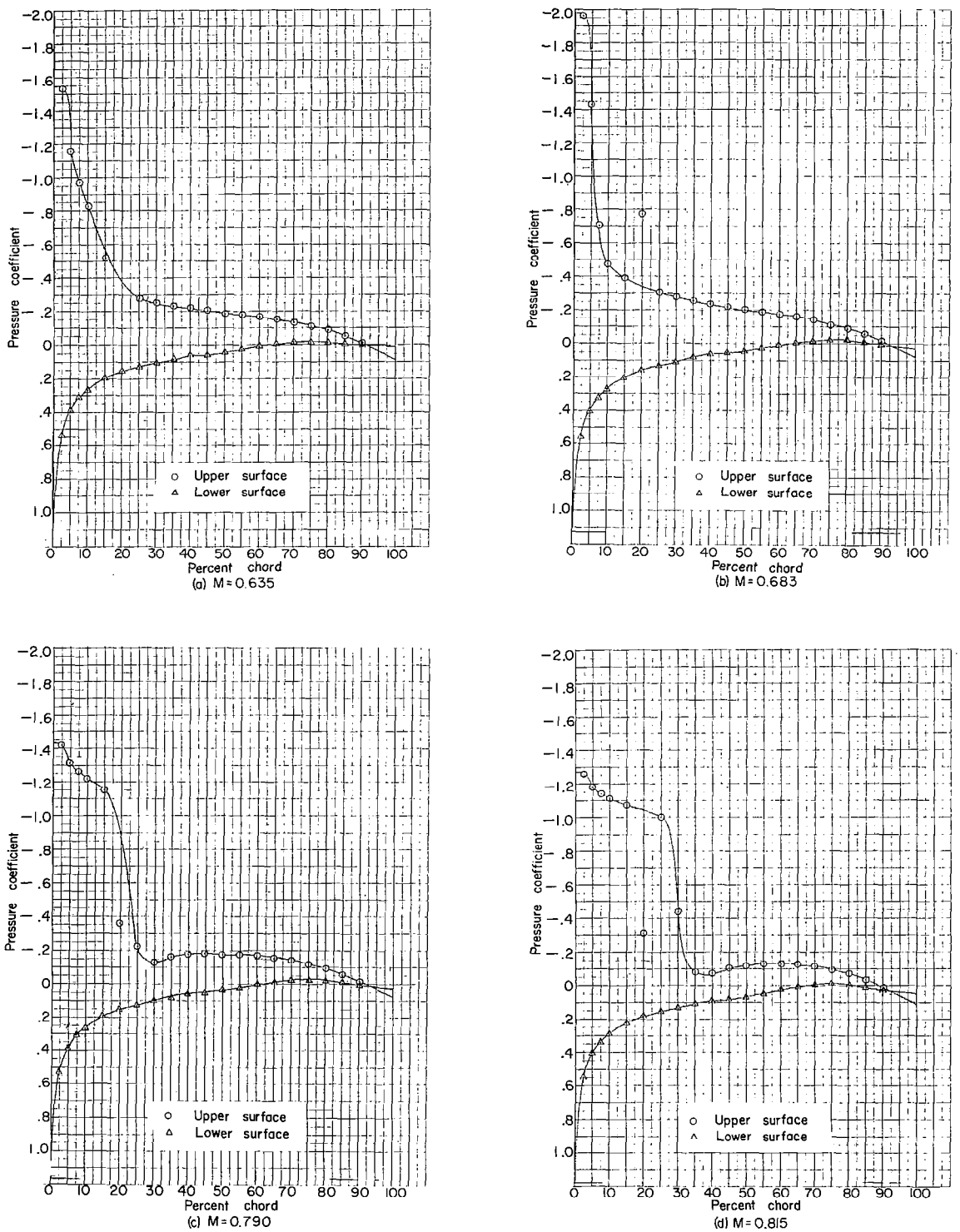
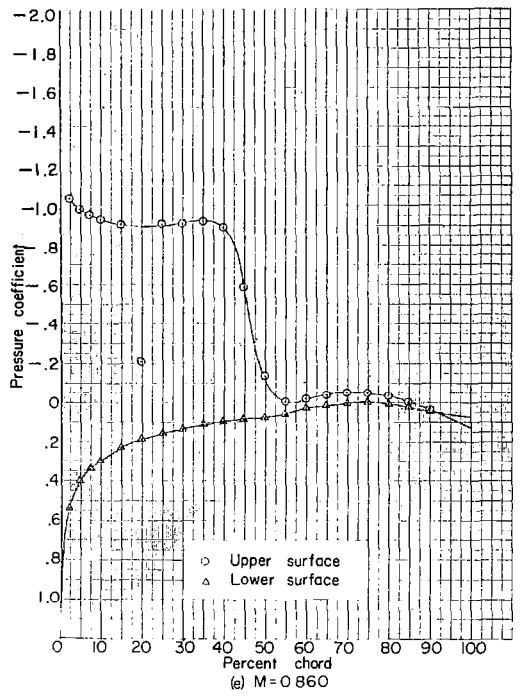
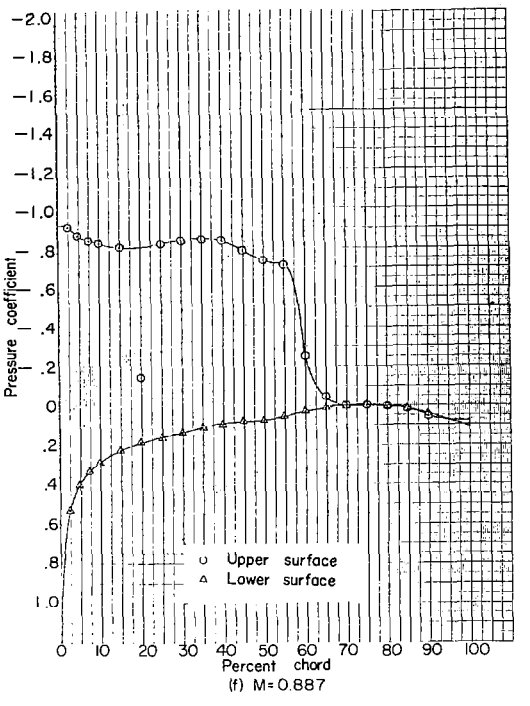
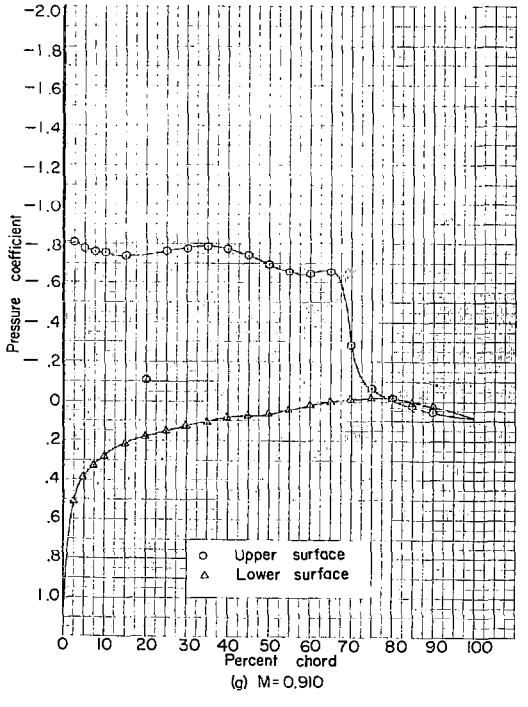


Figure 7.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 4^{\circ}$.







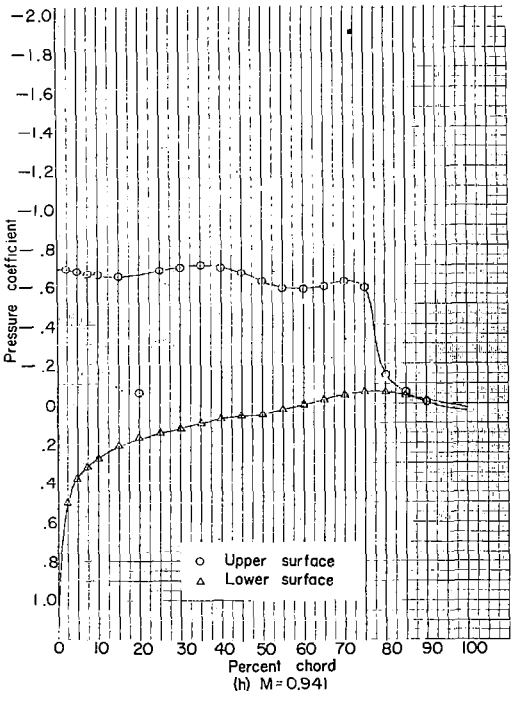
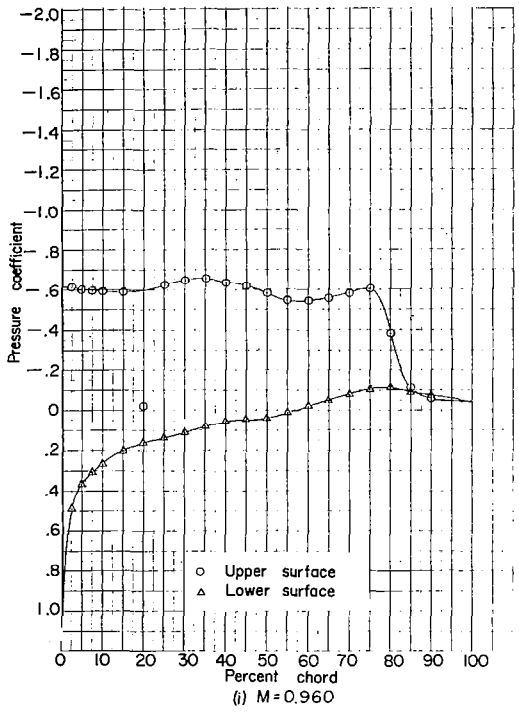
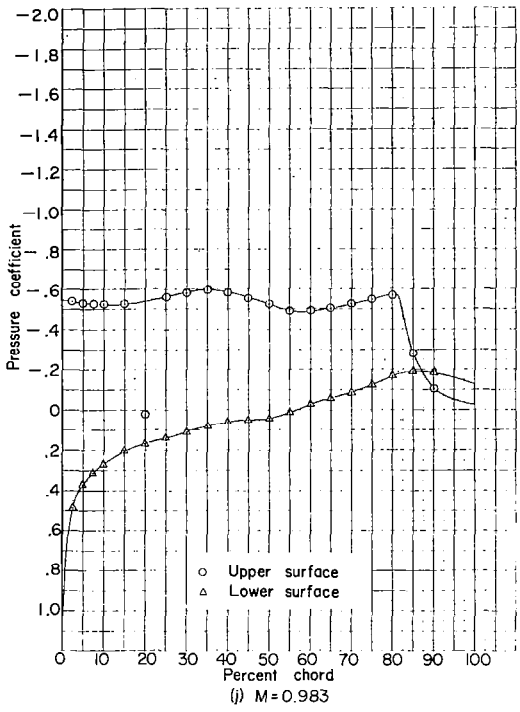


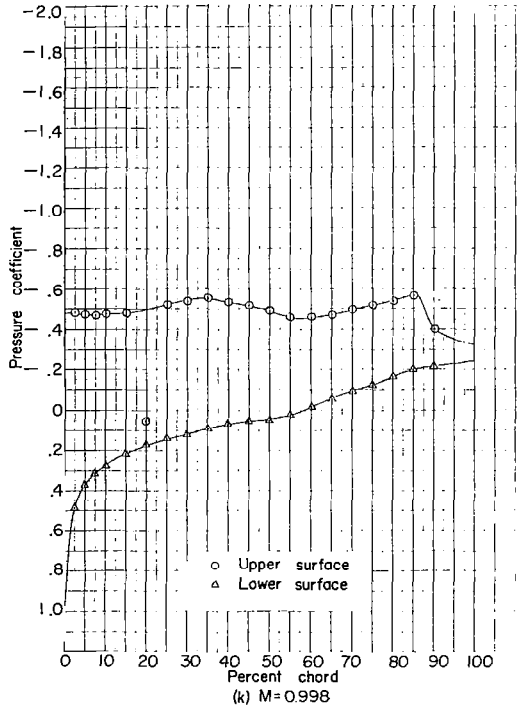
Figure 7.- Continued. NACA 16-004; $\alpha = 4^{\circ}$.



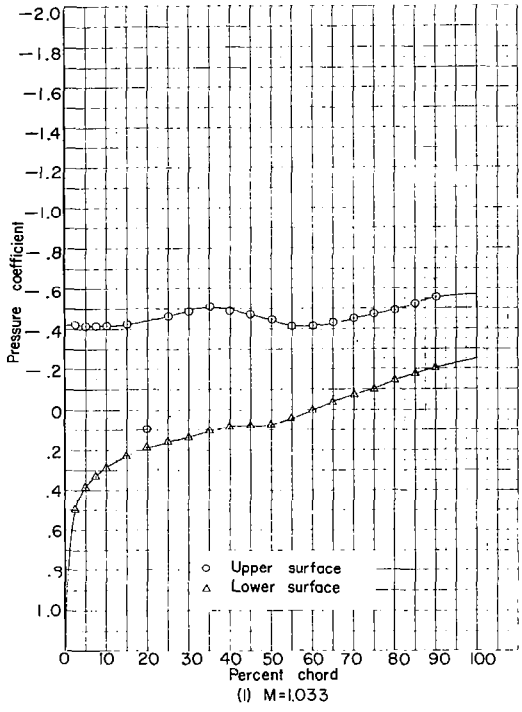
(i) M = 0.960



(j) M = 0.983

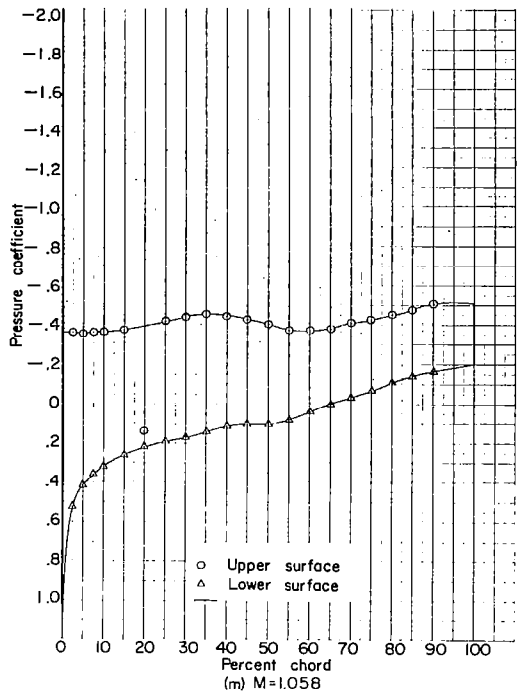


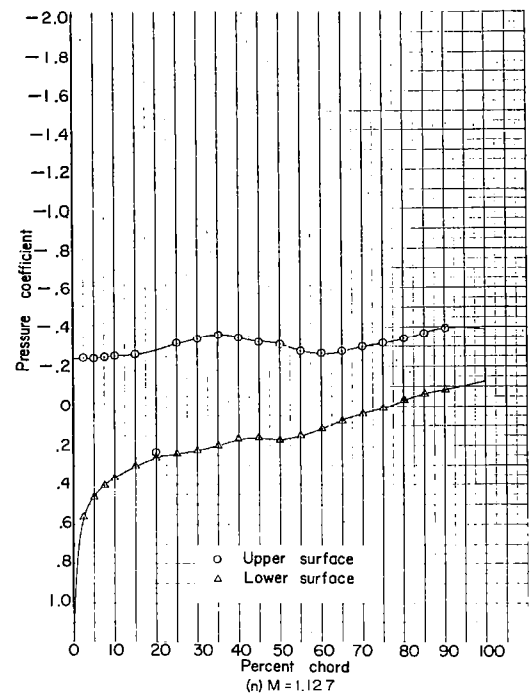
(k) M = 0.998

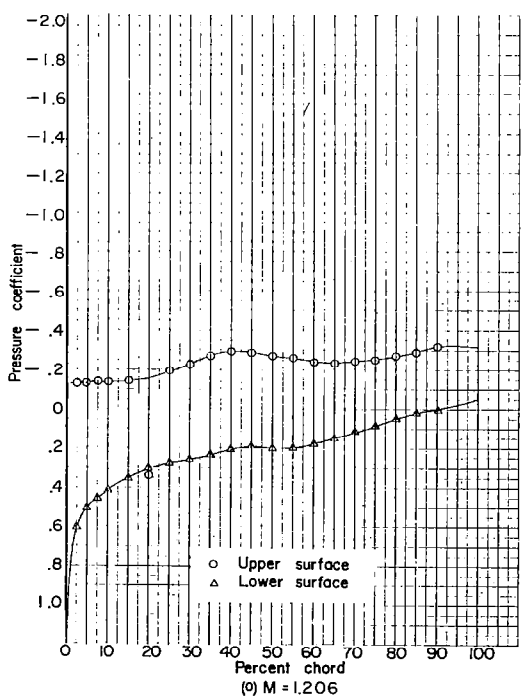


(l) M = 1.033

Figure 7.- Continued. NACA 16-004; $\alpha = 4^{\circ}$.







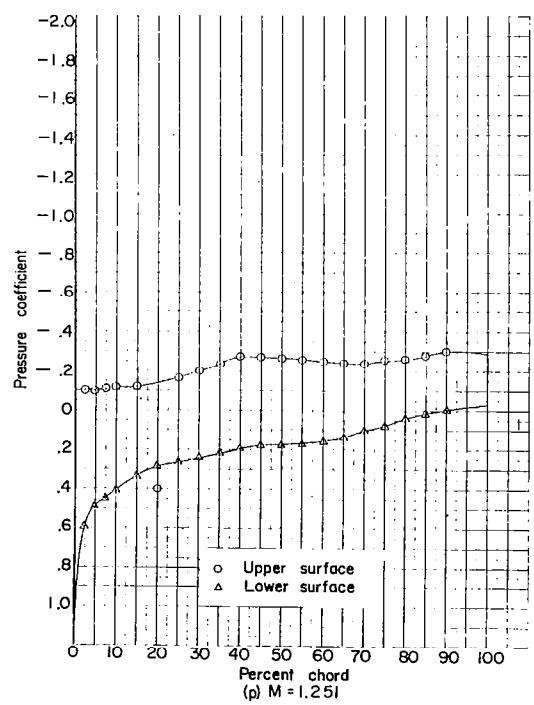


Figure 7.- Concluded. NACA 16-004; $\alpha = 4^{\circ}$.

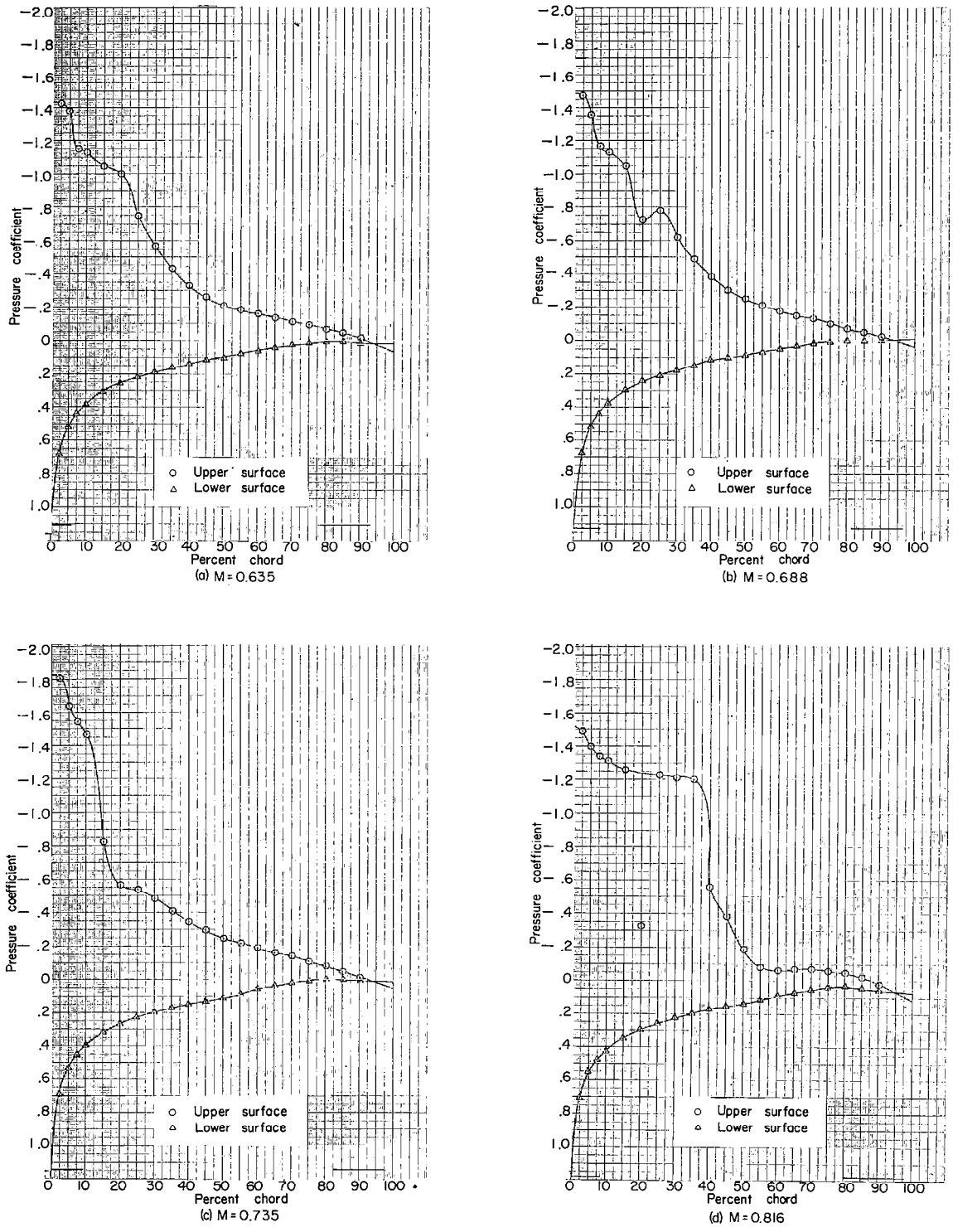
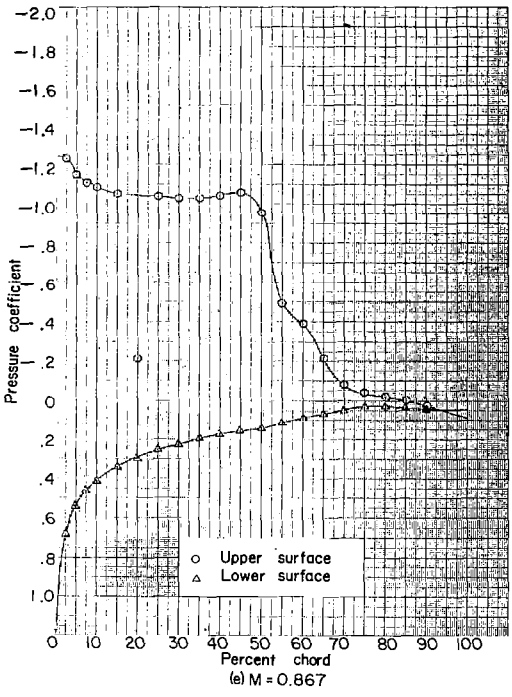
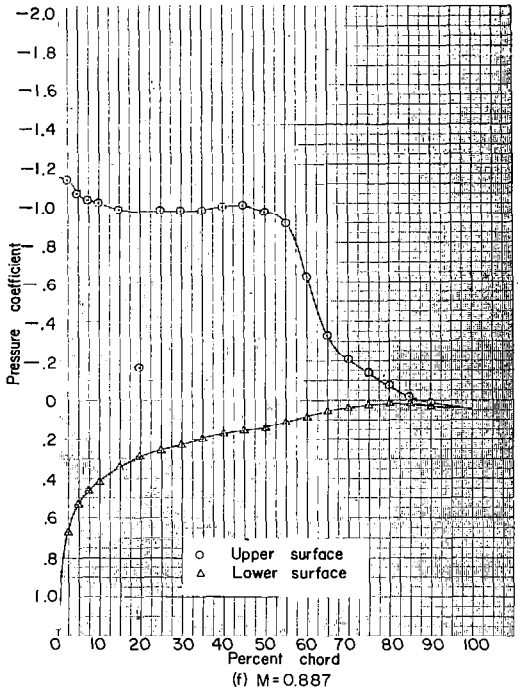
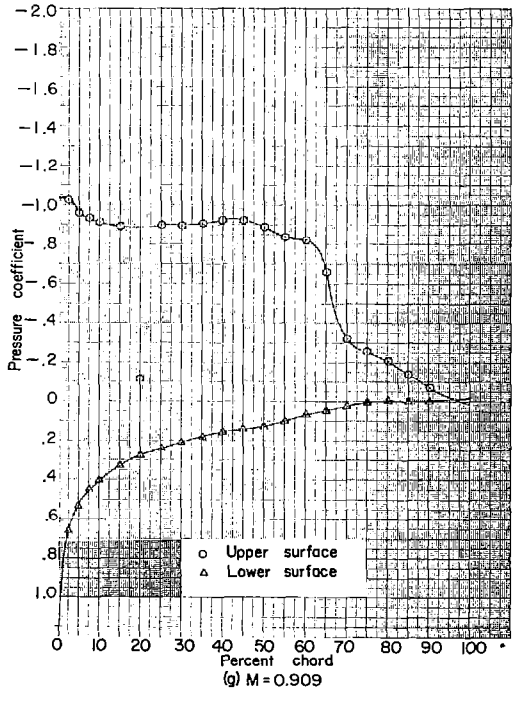


Figure 8.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 6^{\circ}$.







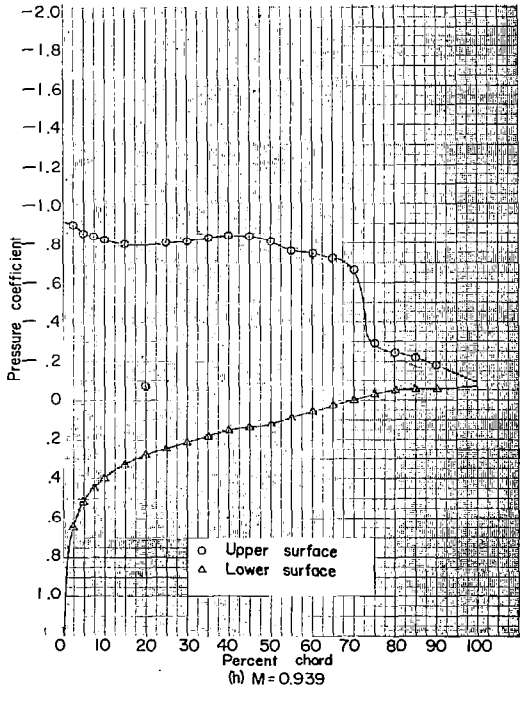
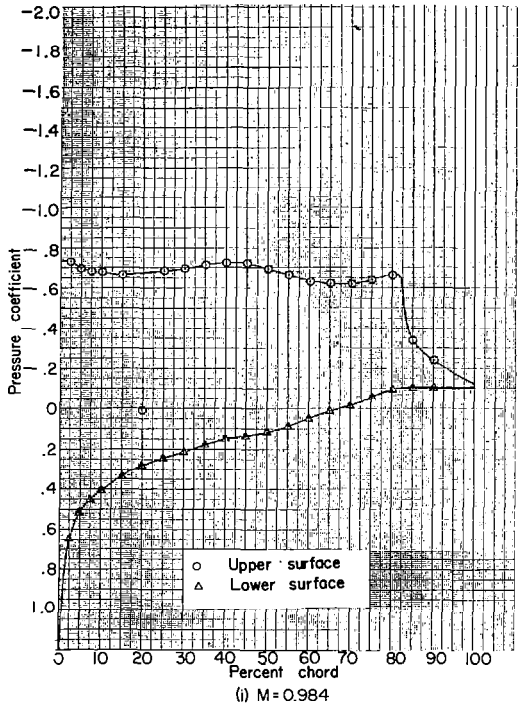
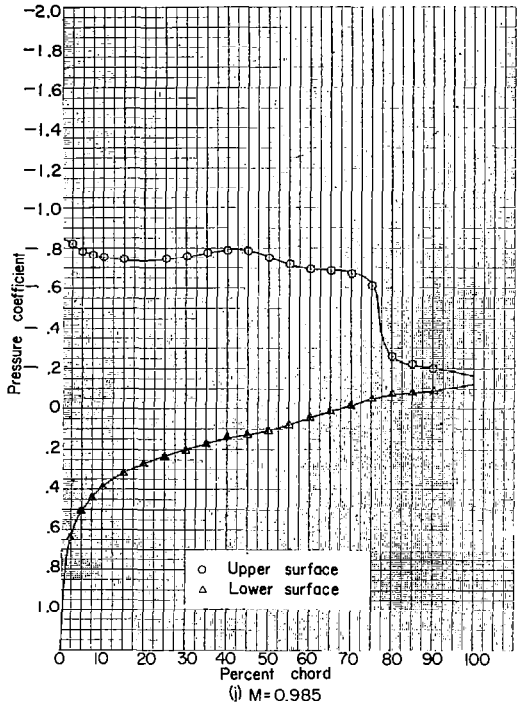
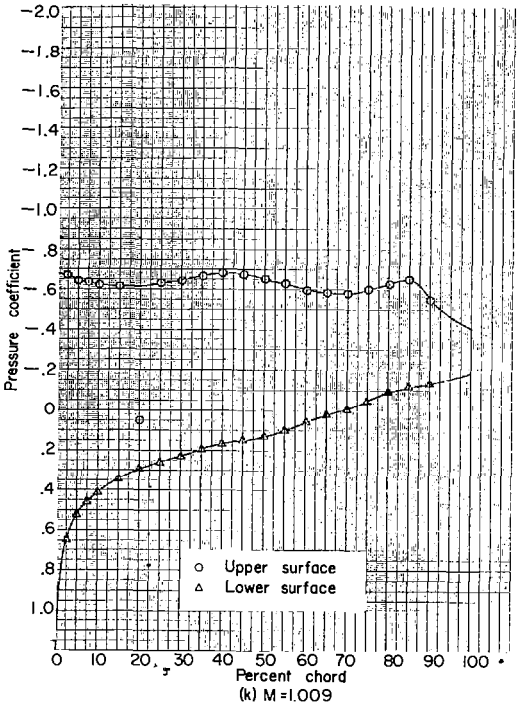


Figure 8.- Continued. NACA 16-004; $\alpha = 6^{\circ}$.







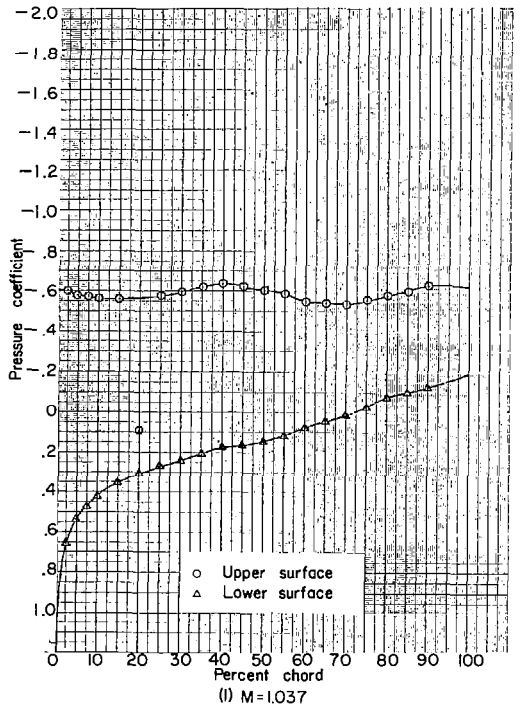
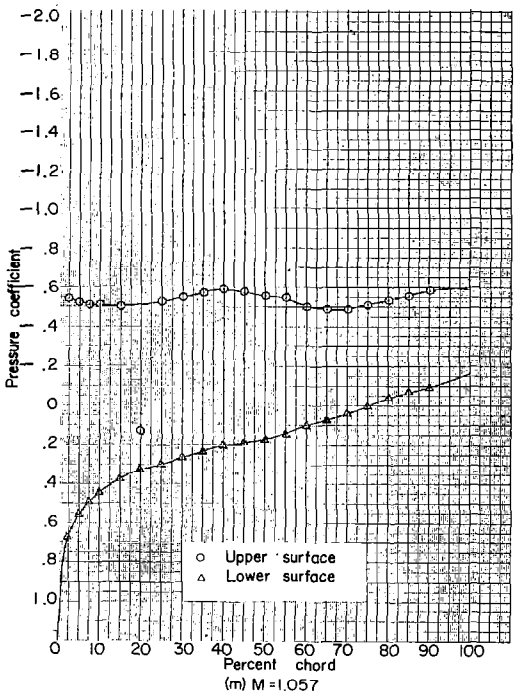
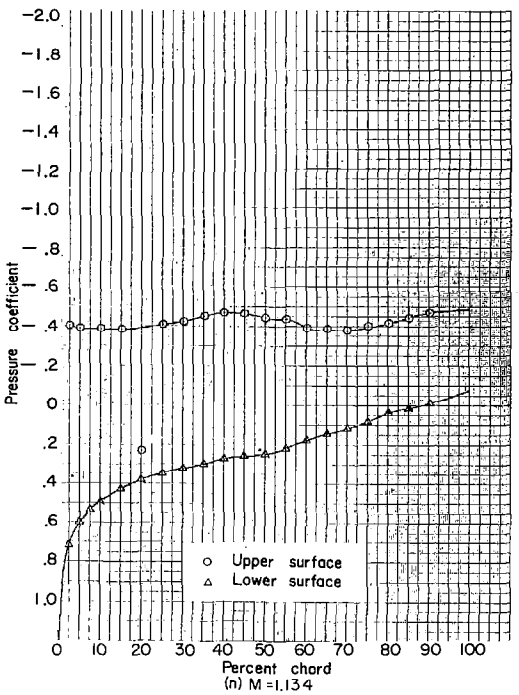
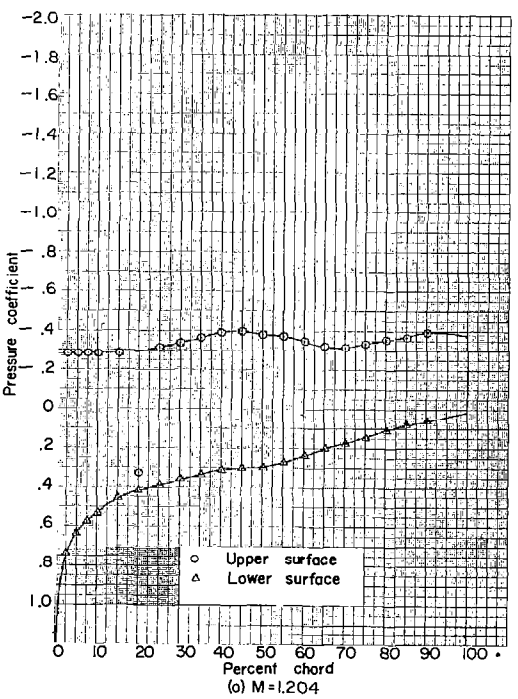


Figure 8.- Continued. NACA 16-004; $\alpha = 6^{\circ}$.







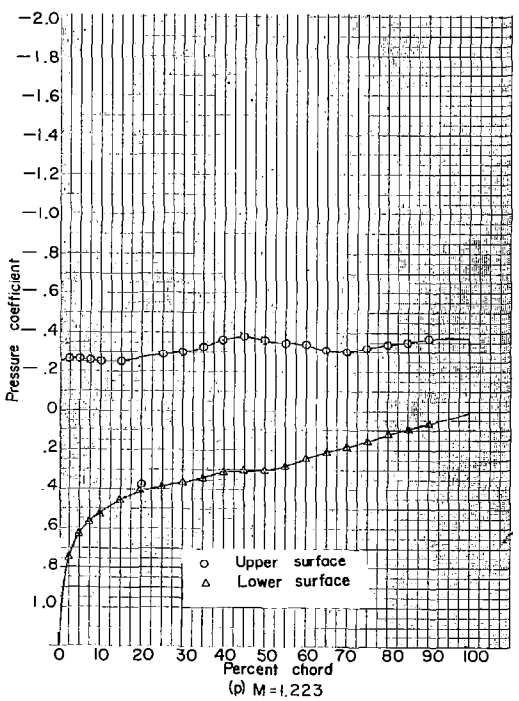


Figure 8.- Concluded. NACA 16-004; $\alpha = 6^{\circ}$.

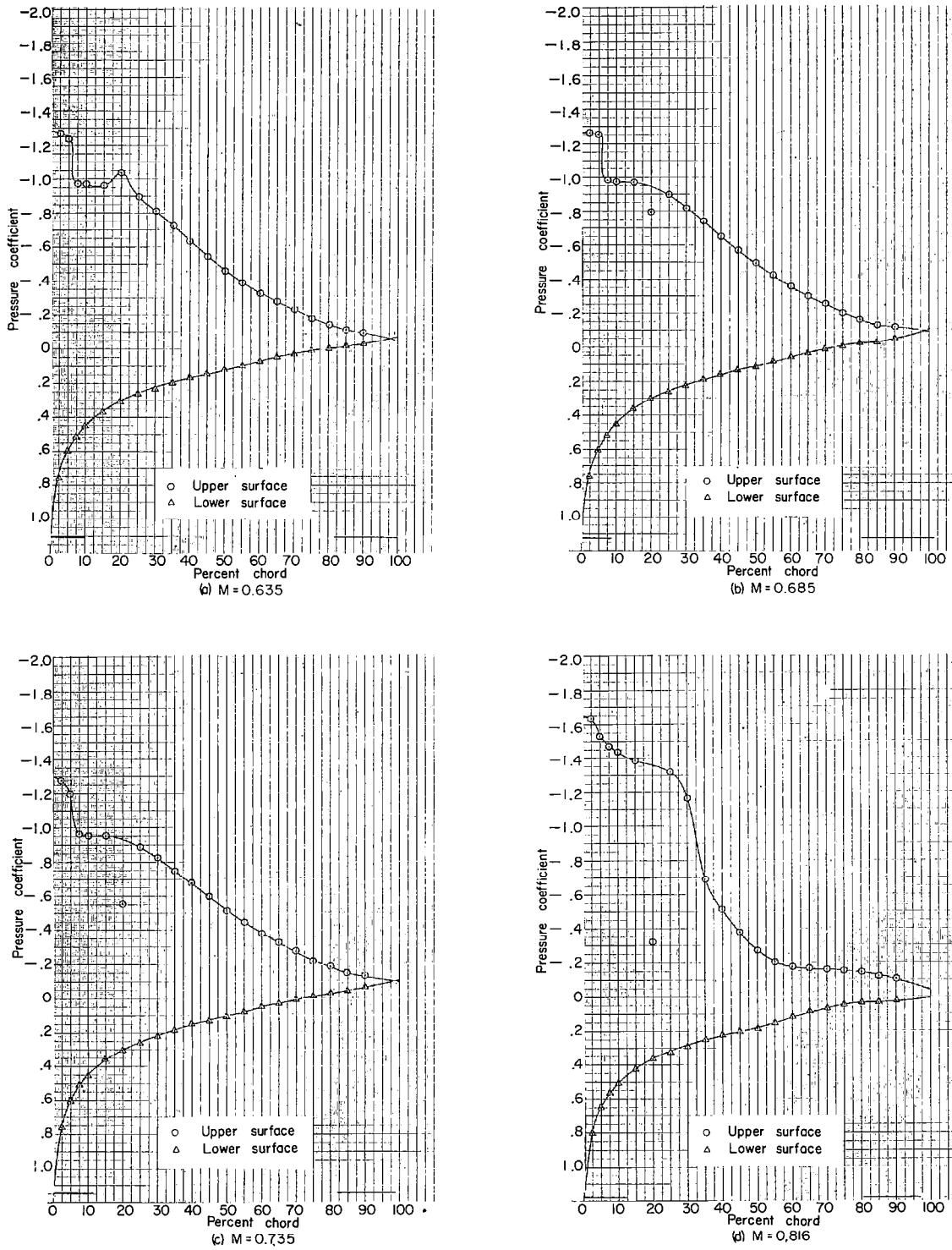
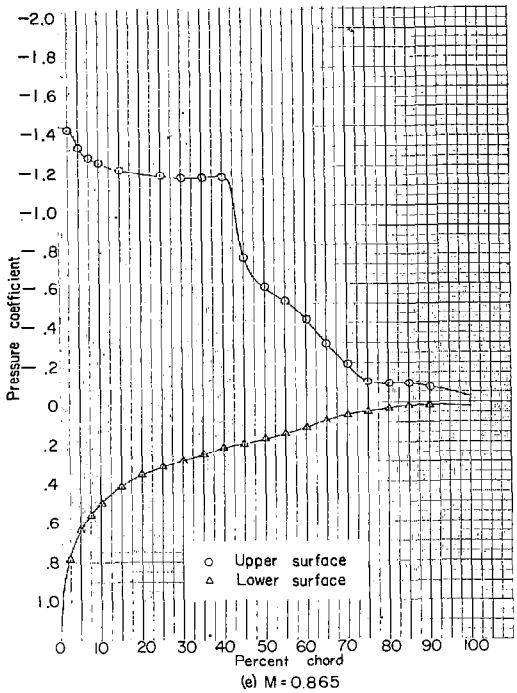
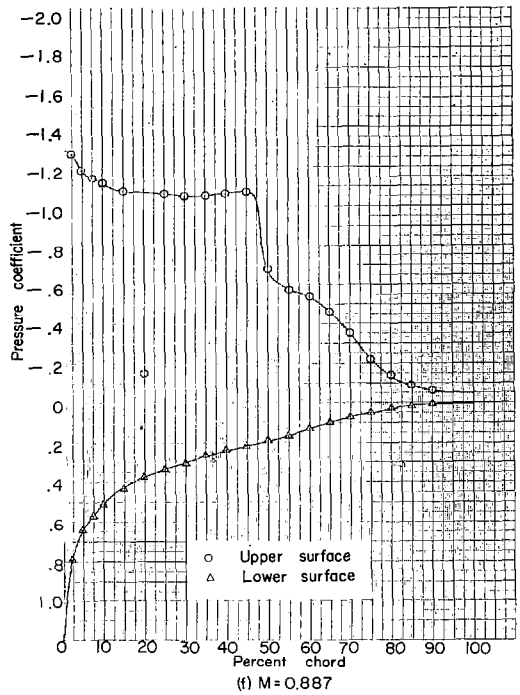


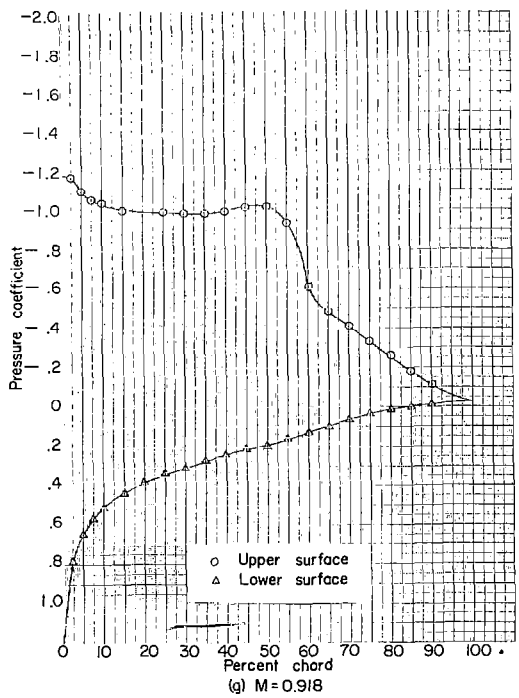
Figure 9.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 8^{\circ}$.



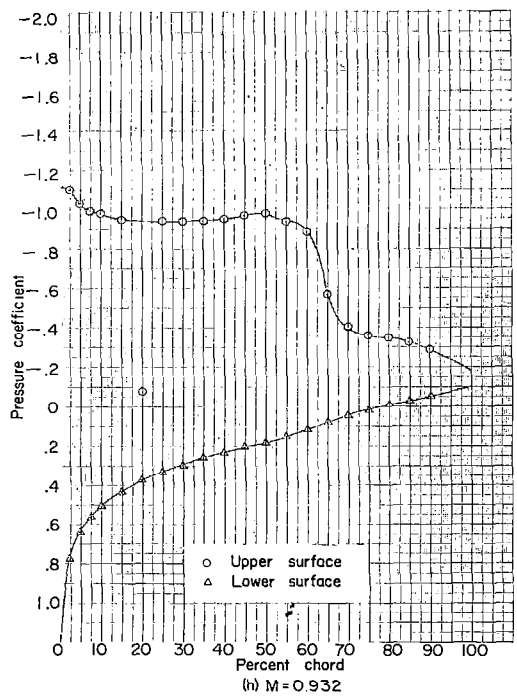
(e) M = 0.865



(f) M = 0.887

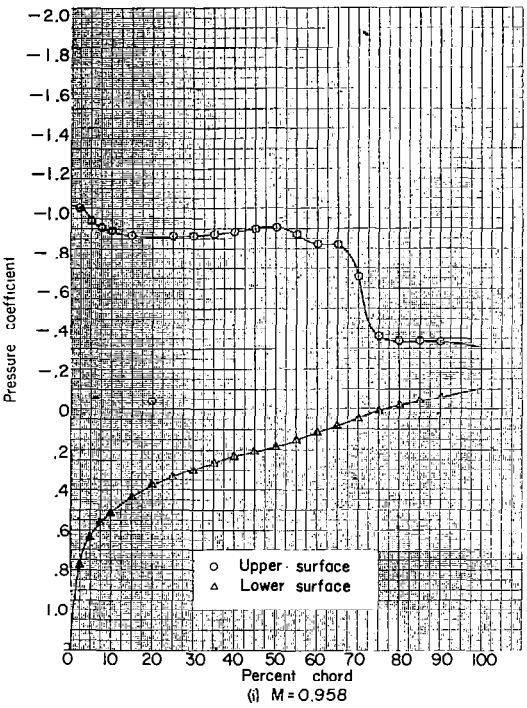


(g) M = 0.918

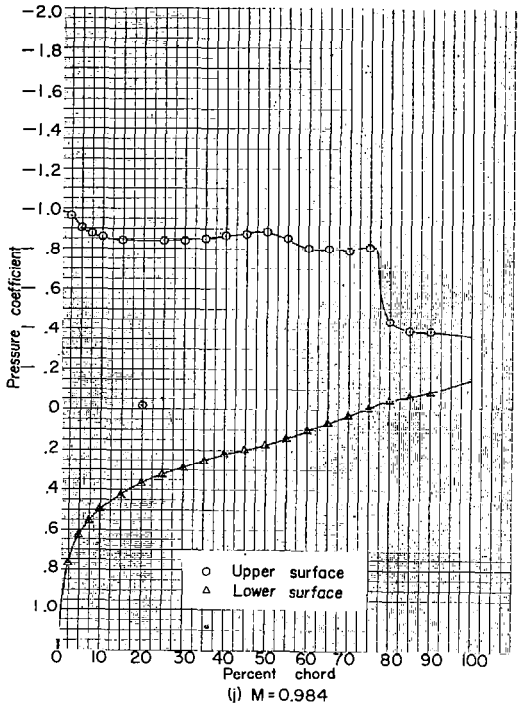


(h) M = 0.932

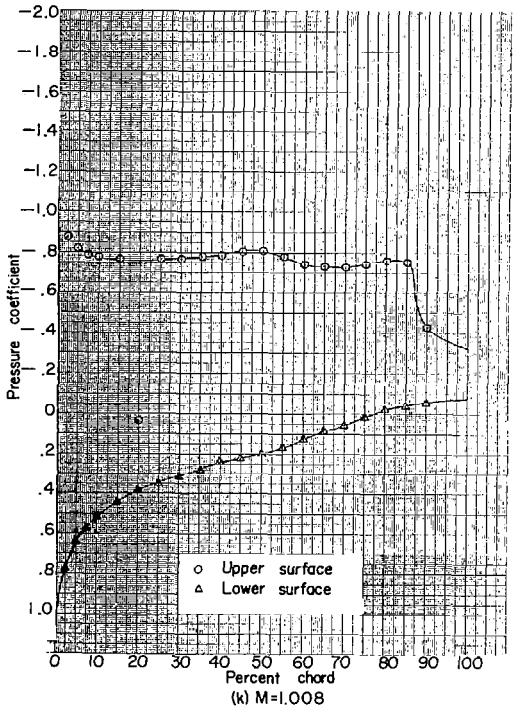
Figure 9.- Continued. NACA 16-004; $\alpha = 8^{\circ}$.



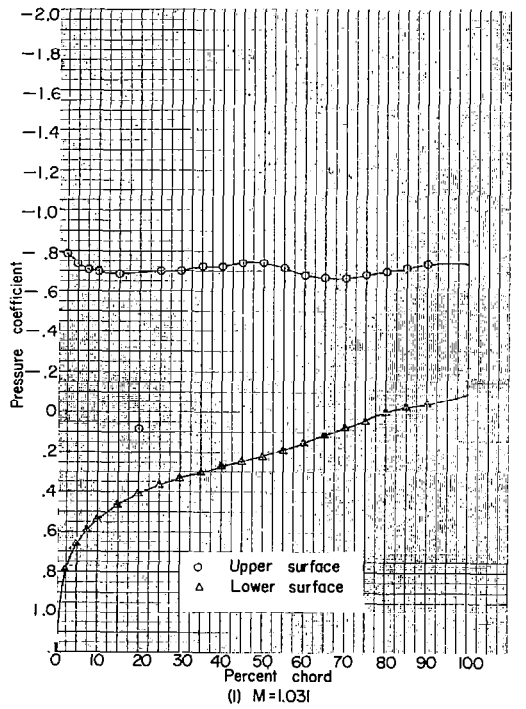
(i) M=0.958



(j) M=0.984

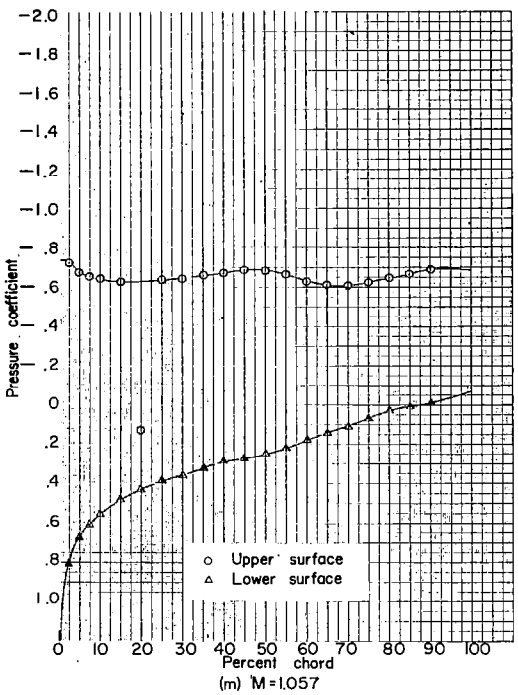


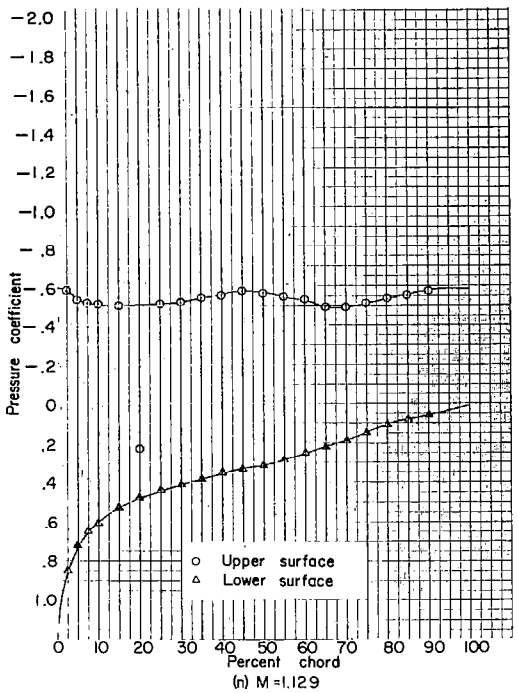
(k) M=1.008

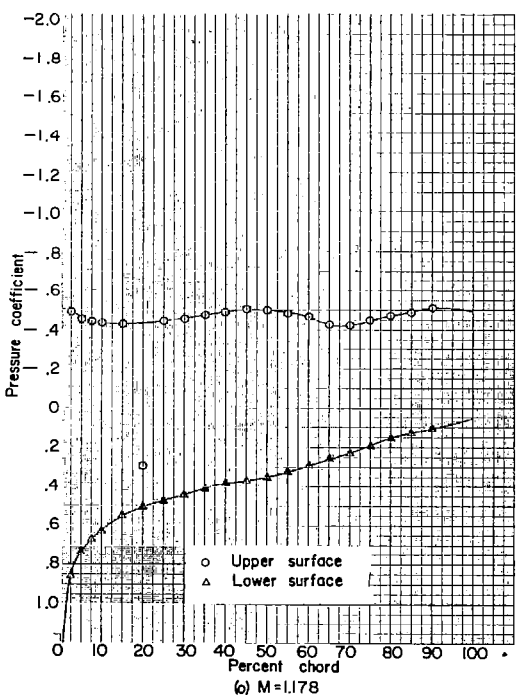


(l) M=1.031

Figure 9.- Continued. NACA 16-004; $\alpha = 8^{\circ}$.







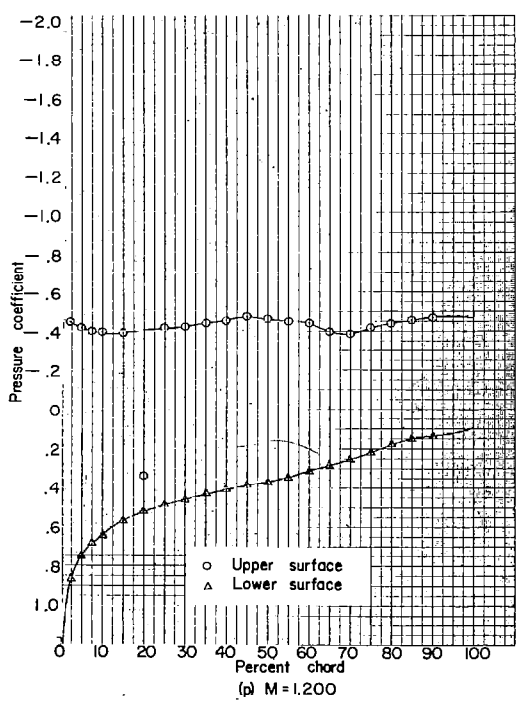


Figure 9.- Concluded. NACA 16-004; $\alpha = 8^{o}$.

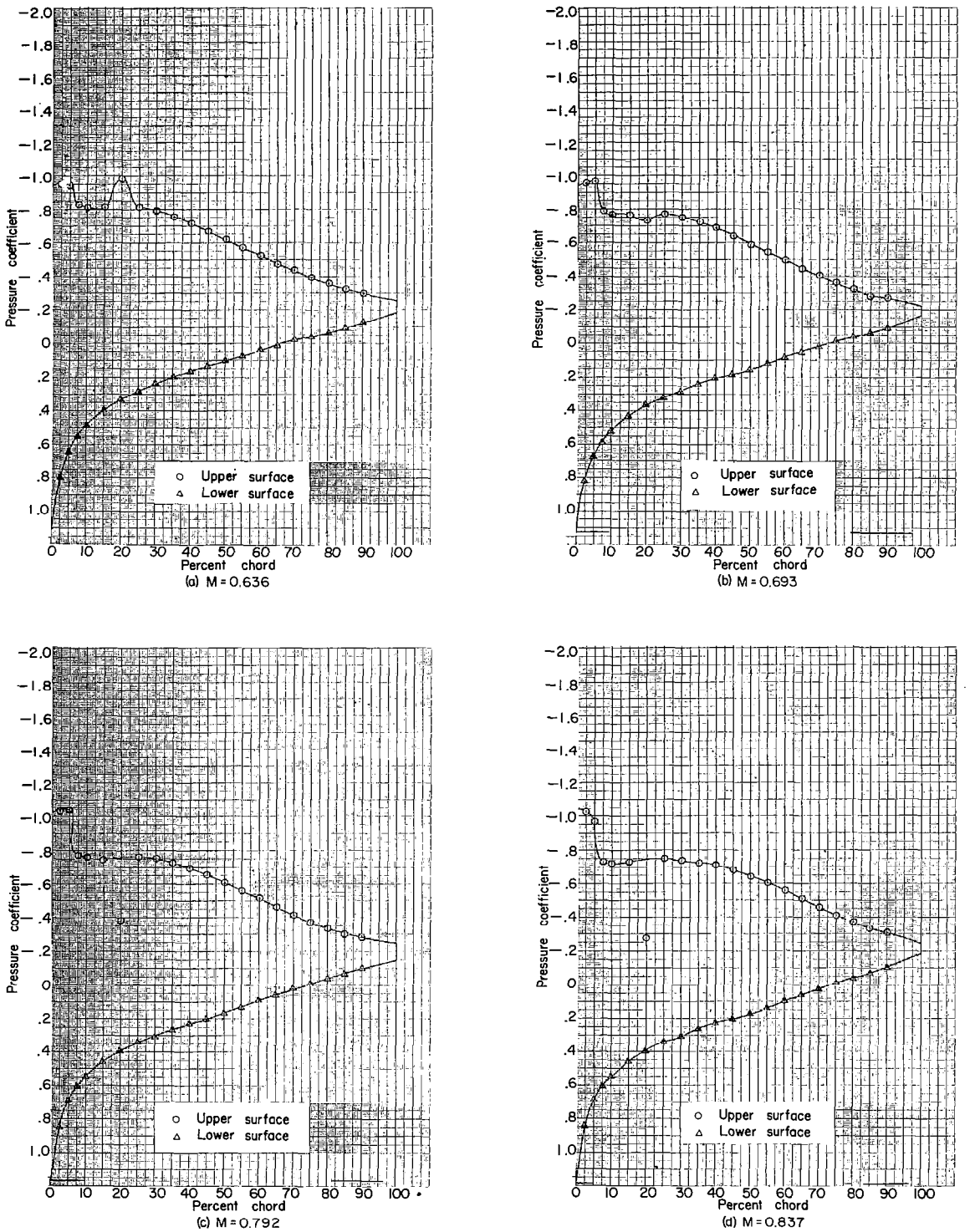
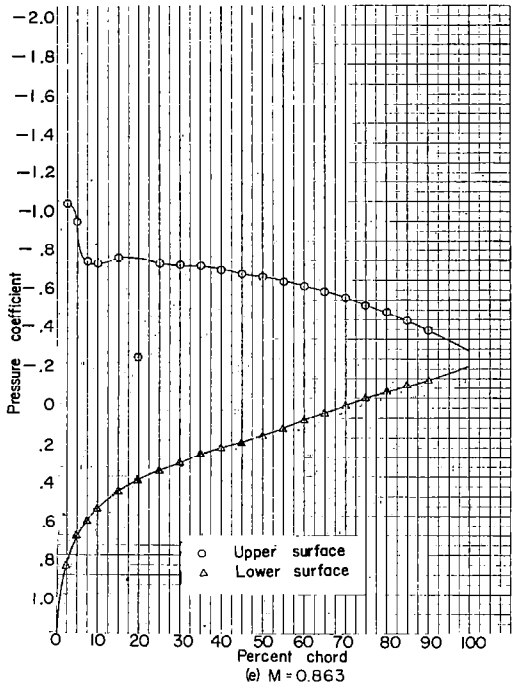
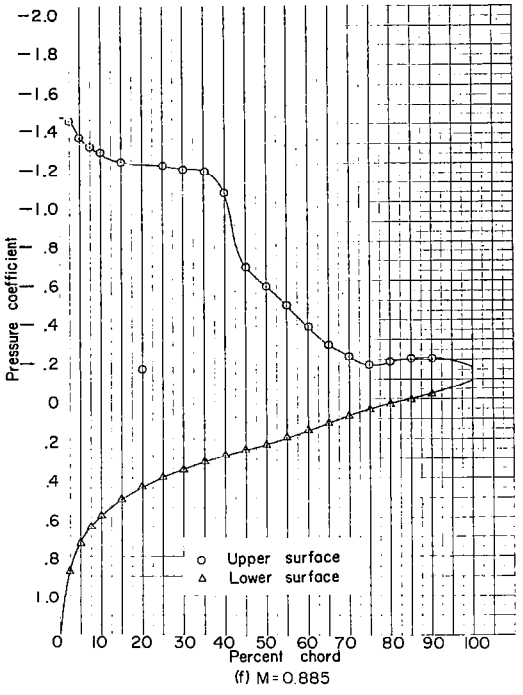
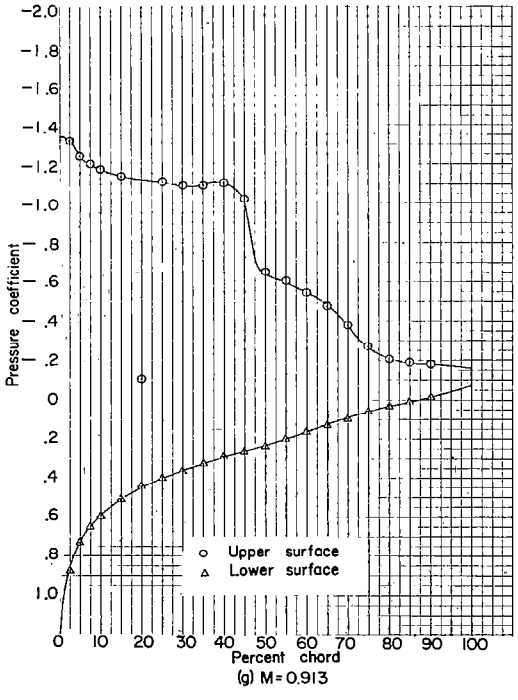


Figure 10.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 10^{\circ}$.







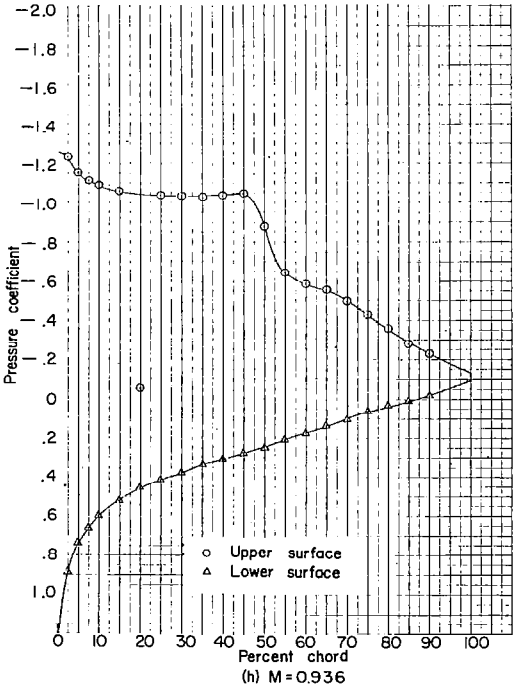
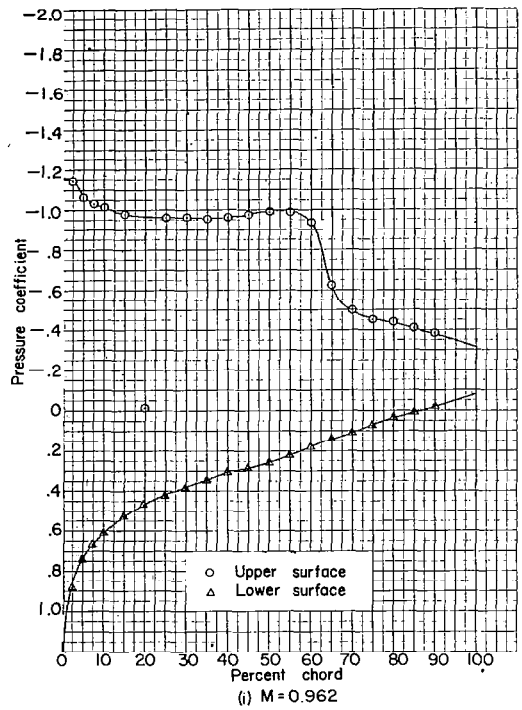
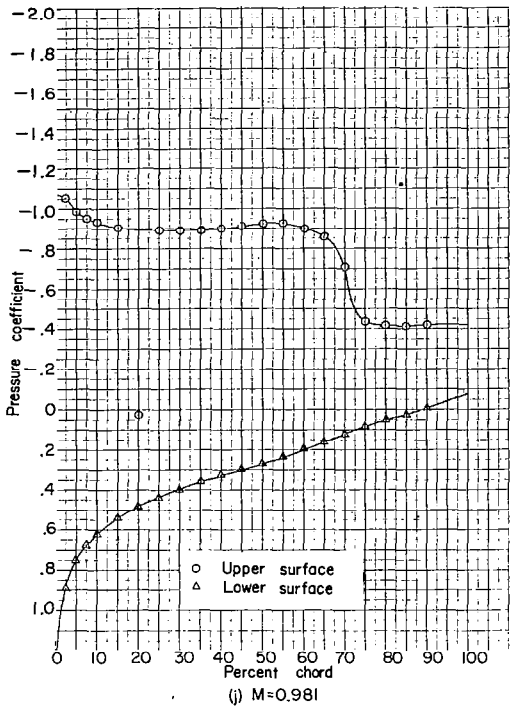


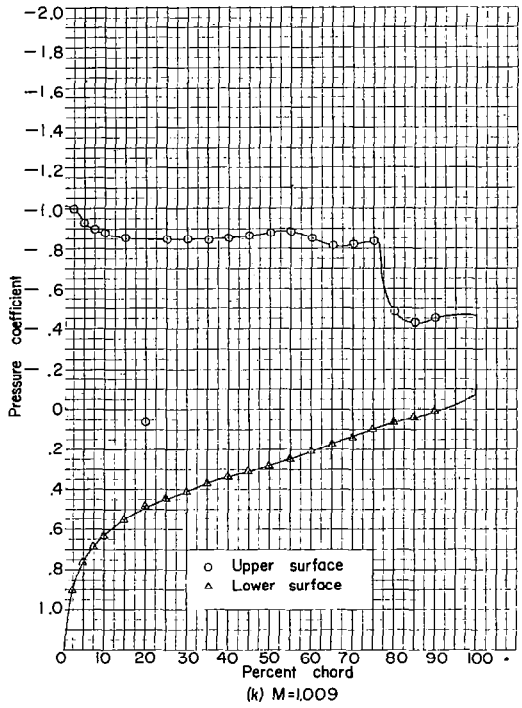
Figure 10.- Continued. NACA 16-004; $\alpha = 10^{\circ}$.



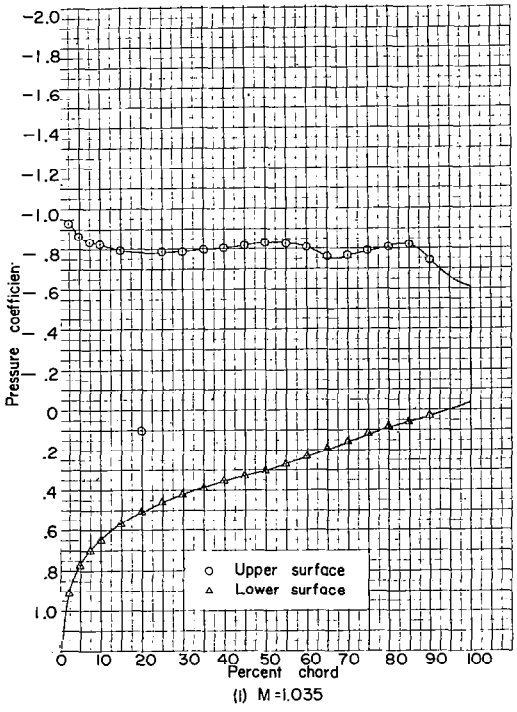
(i) M = 0.962



(j) M=0.981

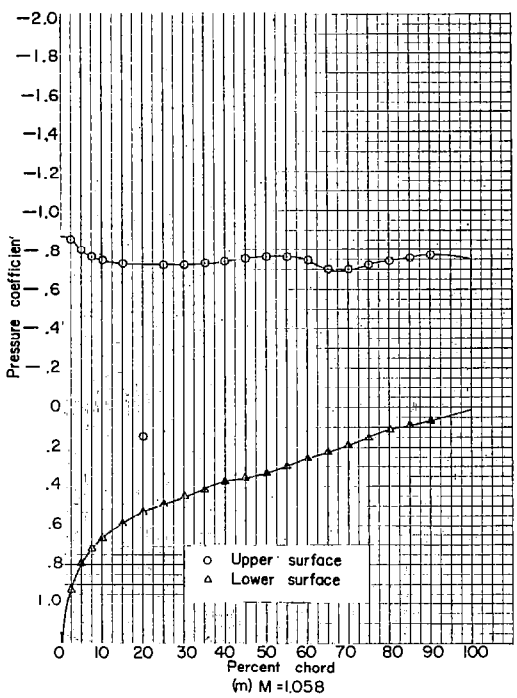


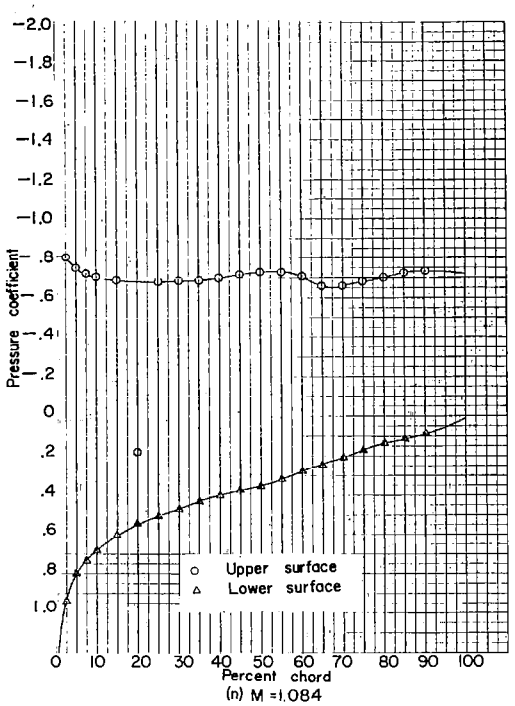
(k) M=1.009

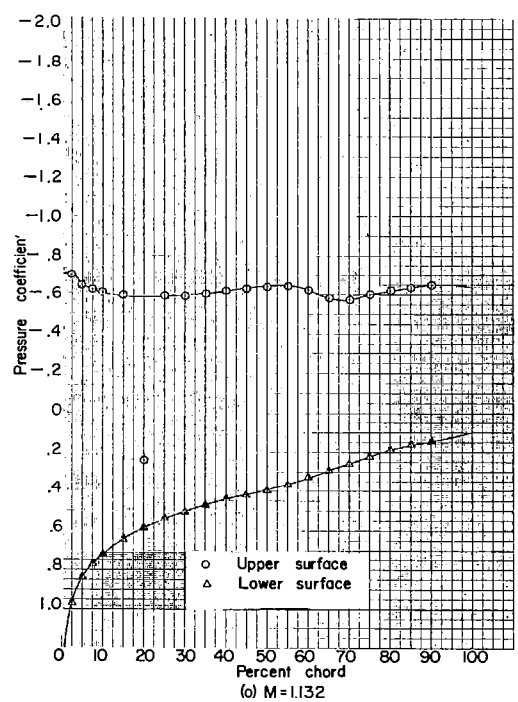


(l) M=1.035

Figure 10.- Continued. NACA 16-004; $\alpha = 10^{o}$.







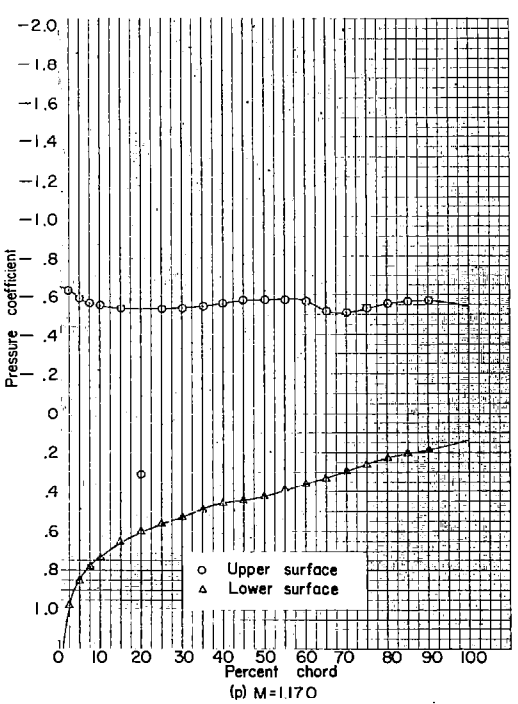


Figure 10.- Concluded. NACA 16-004; $\alpha = 10^{\circ}$.

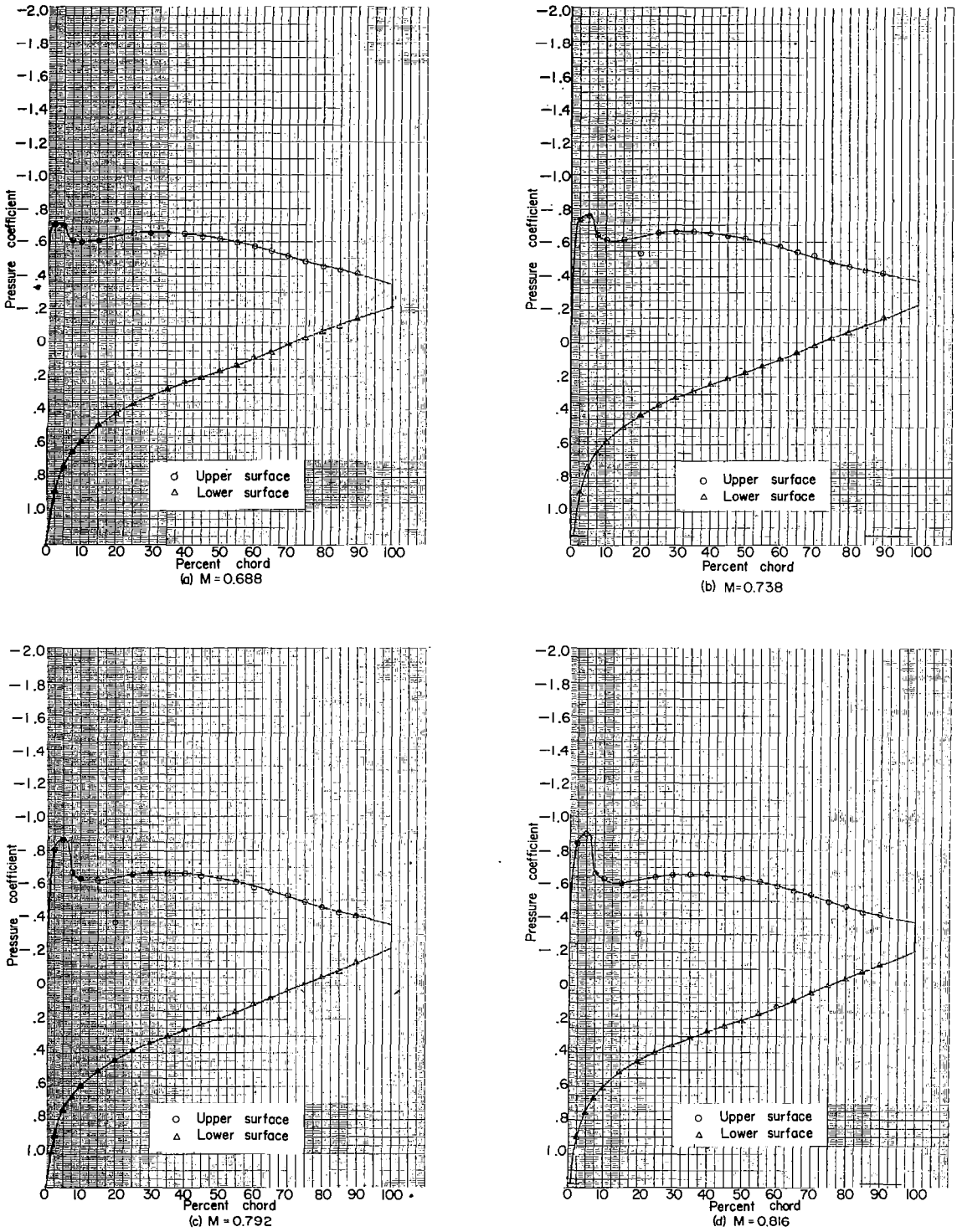
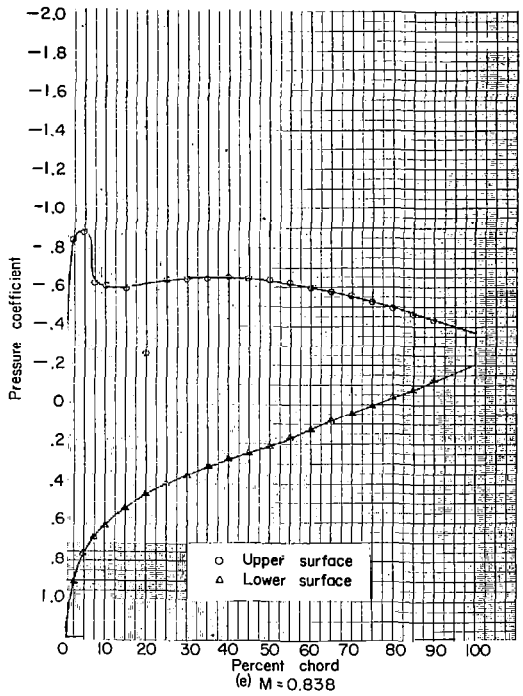
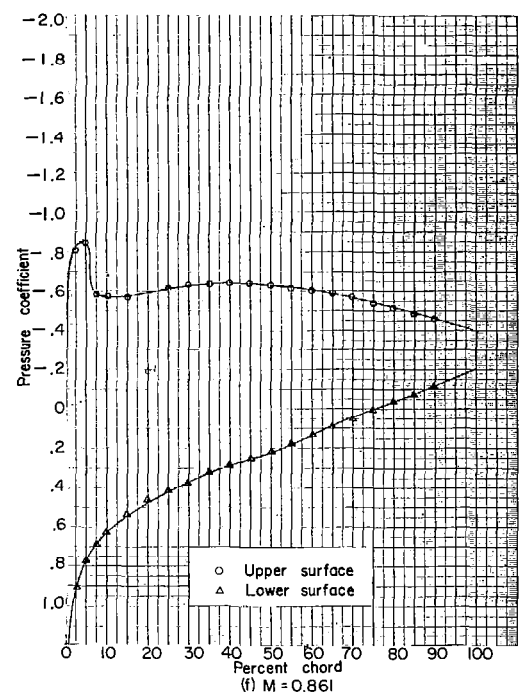
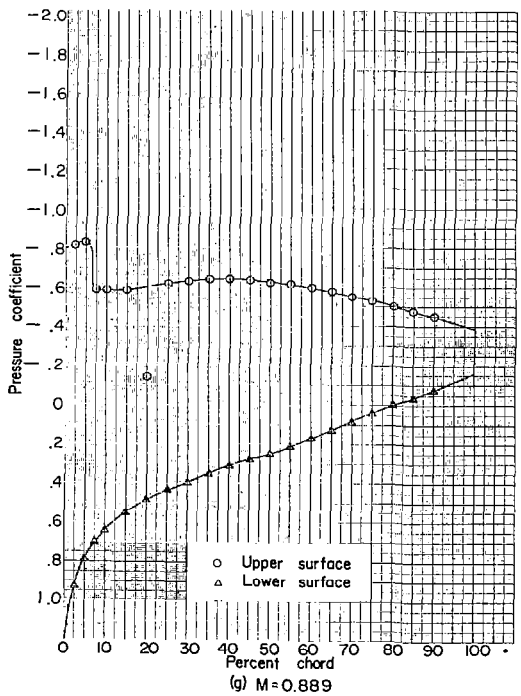


Figure 11.- Pressure distributions over NACA 16-004 airfoil section. $\alpha = 12^{\circ}$.







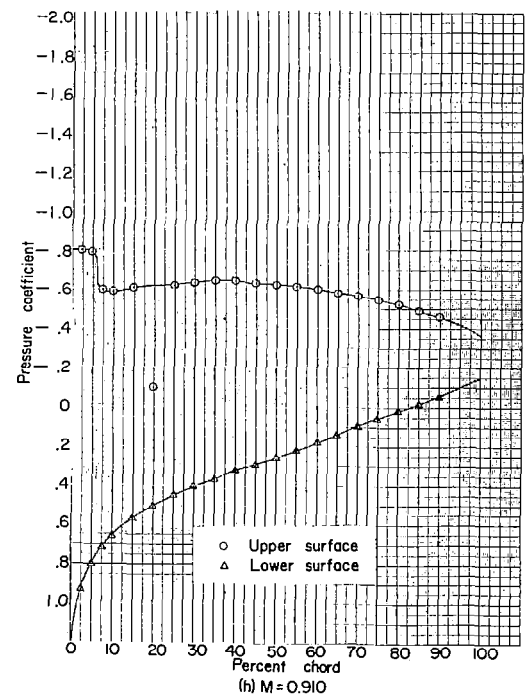
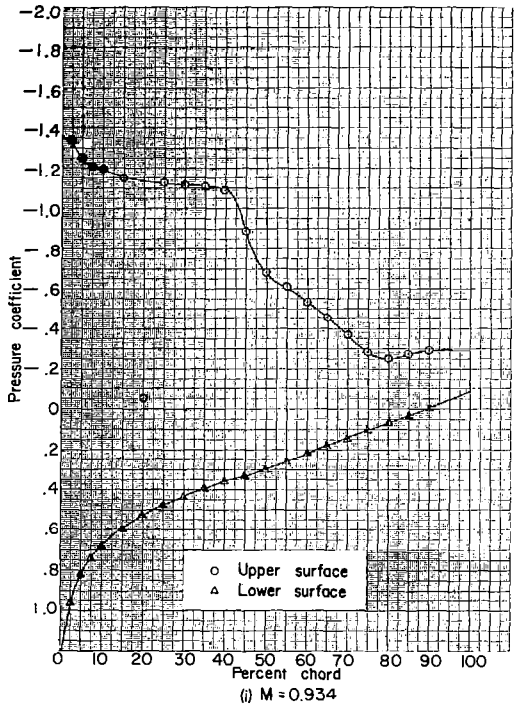
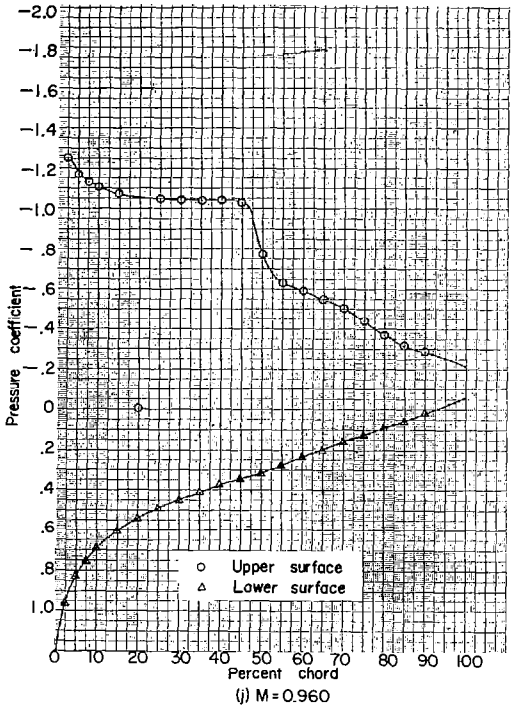


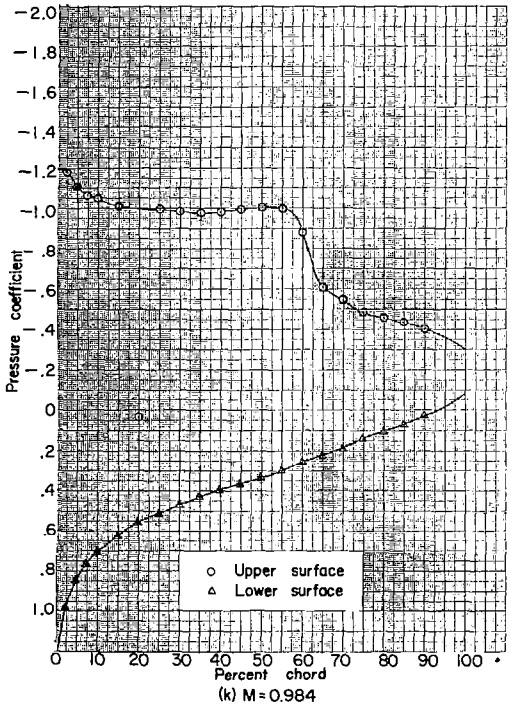
Figure 11.- Continued. NACA 16-004; $\alpha = 12^{\circ}$.



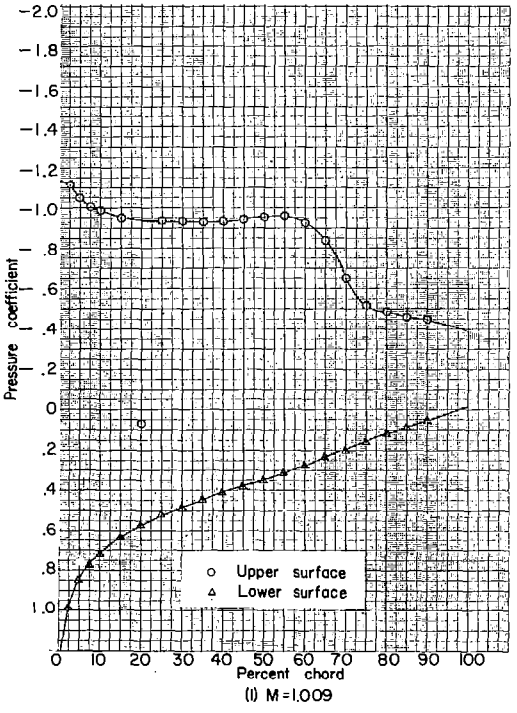
(i) M = 0.934



(j) M = 0.960

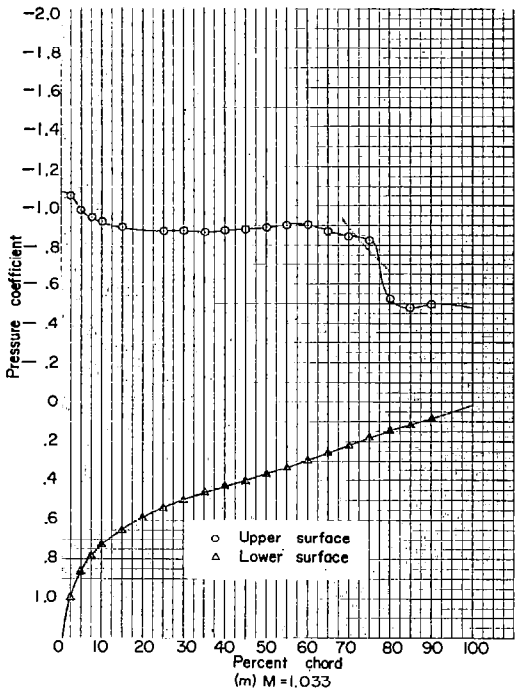


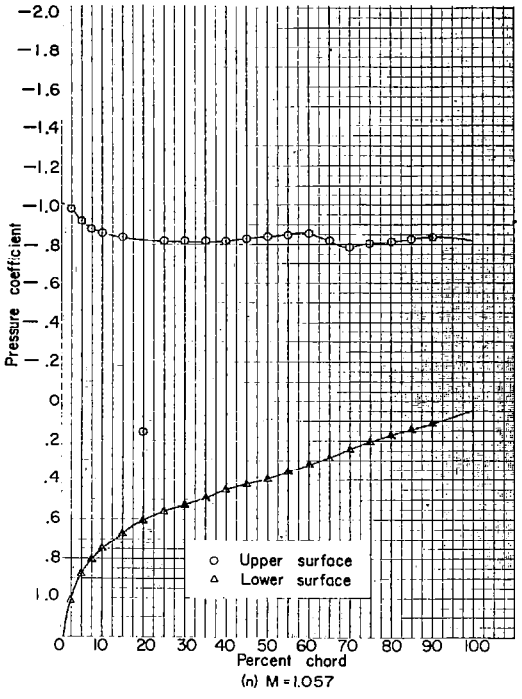
(k) M = 0.984

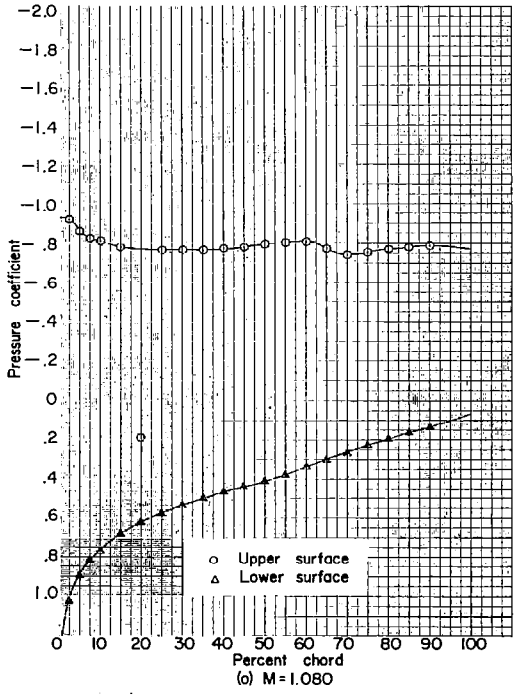


(l) M = 1.009

Figure 11.- Continued. NACA 16-004; $\alpha = 12^{\circ}$.







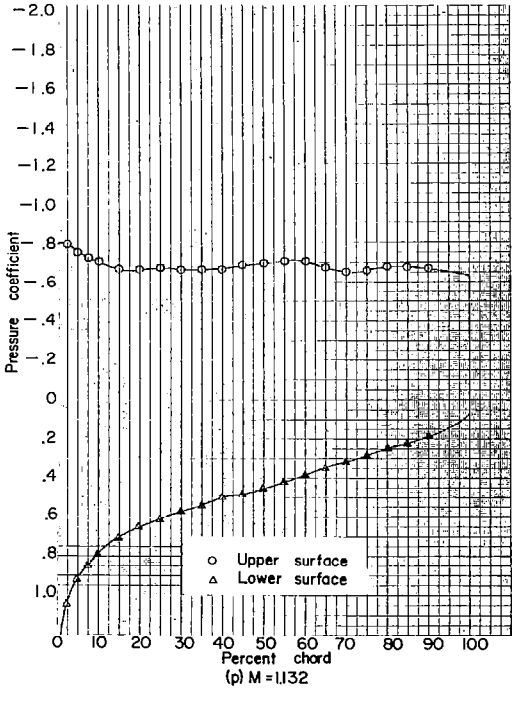


Figure 11.- Concluded. NACA 16-004; $\alpha = 12^{\circ}$.

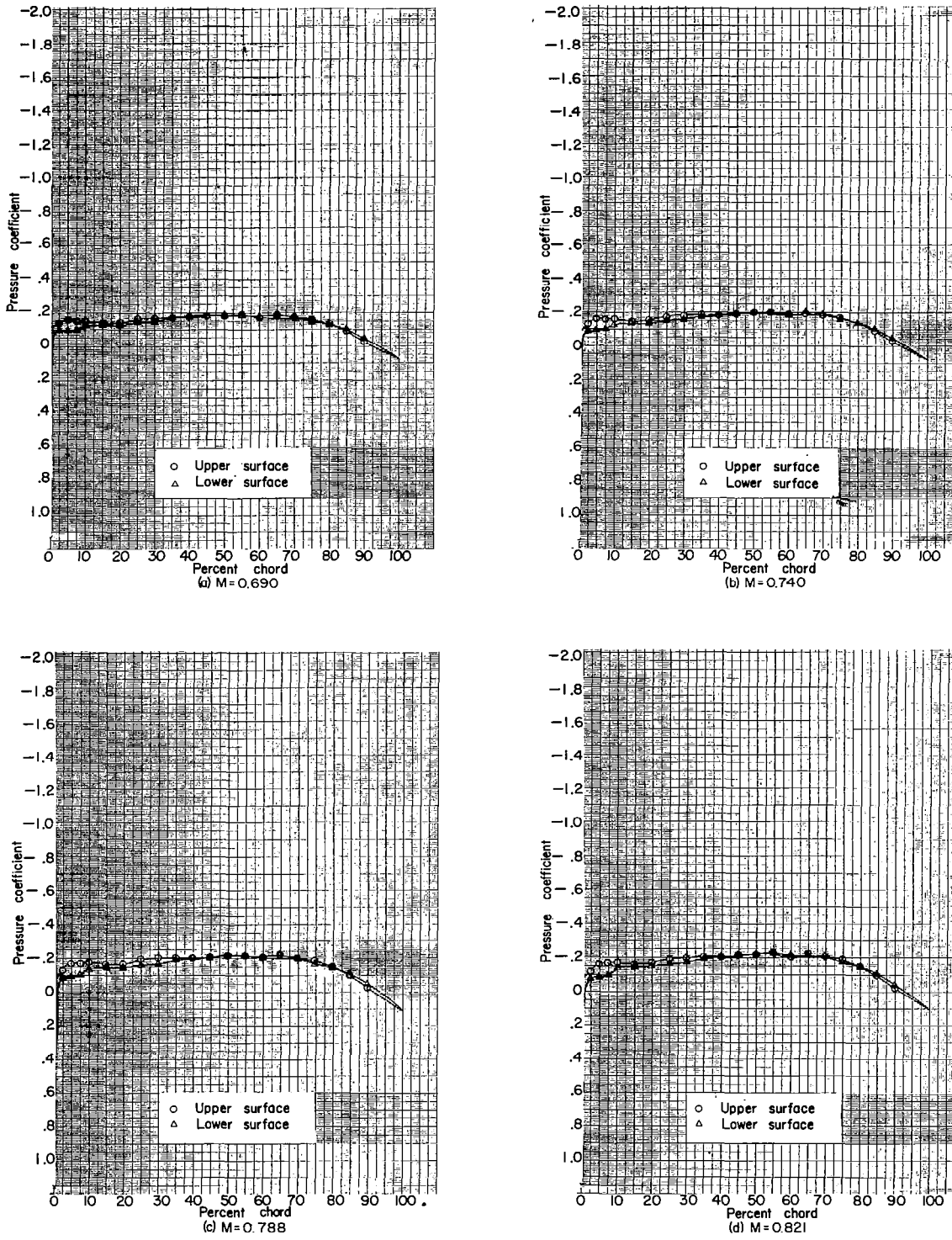
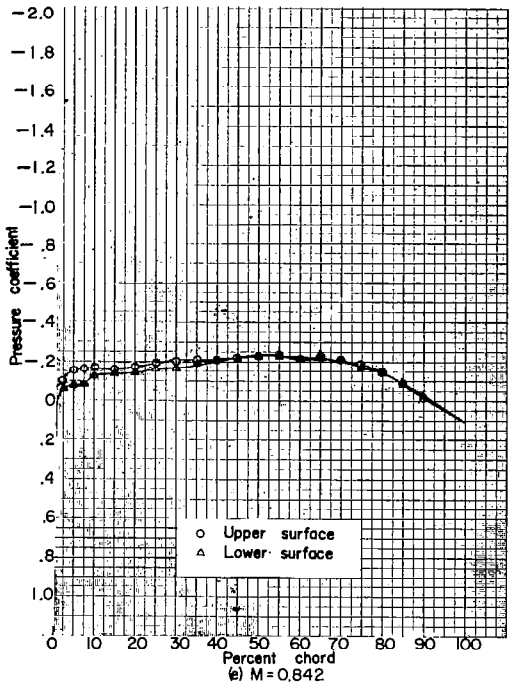
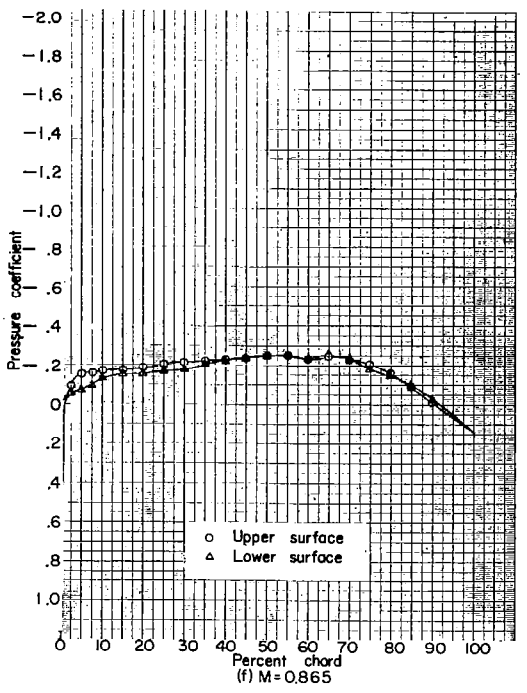
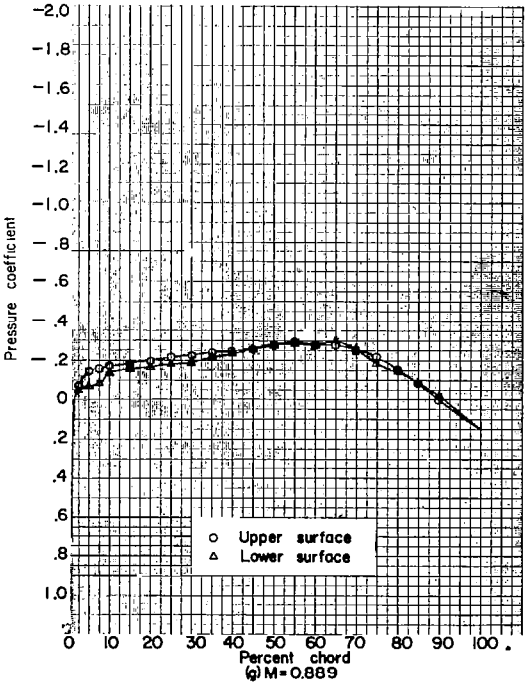


Figure 12.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 0^{\circ}$.







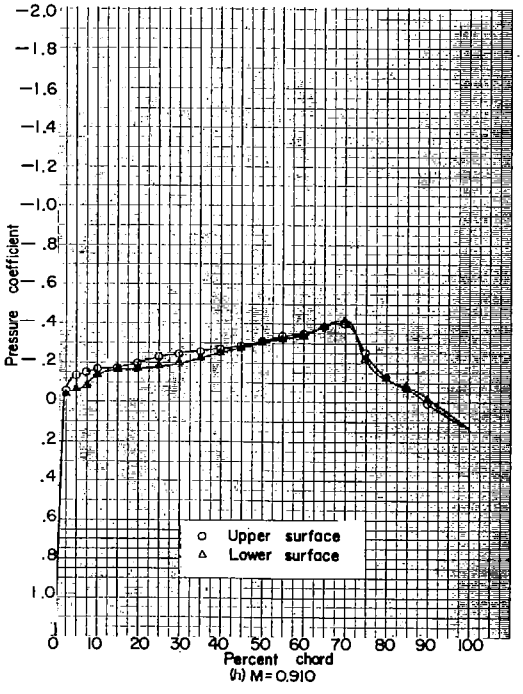
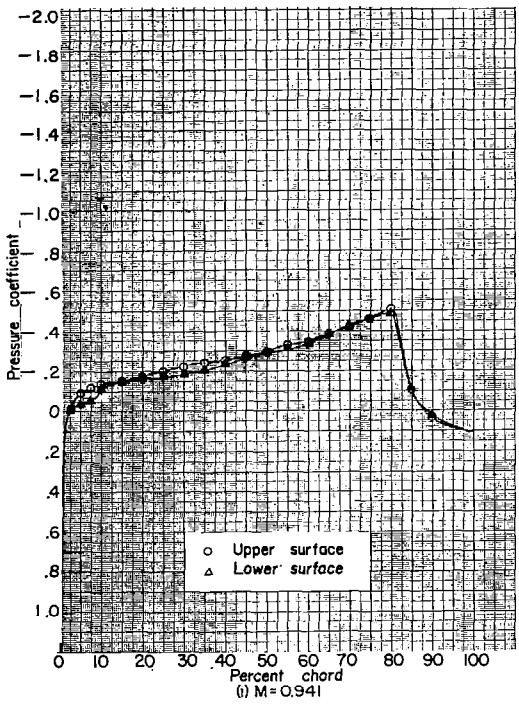
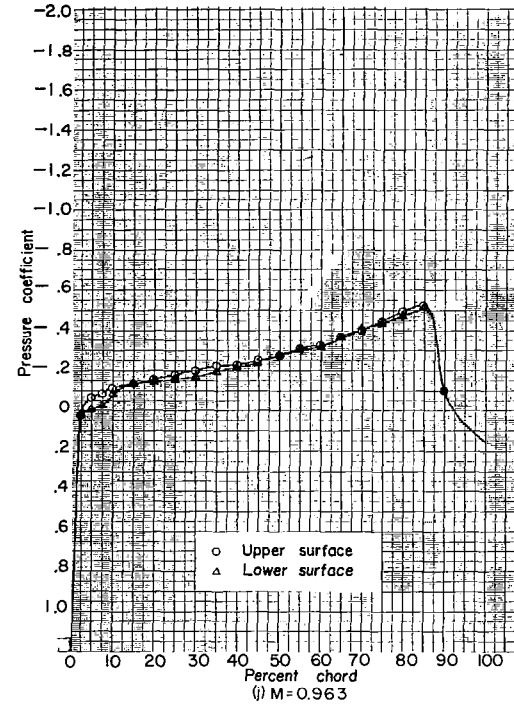
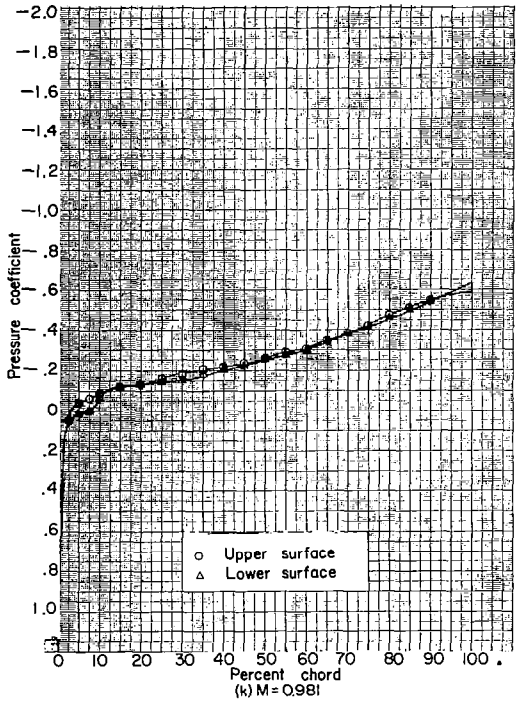


Figure 12.- Continued. NACA 16-006; $\alpha = 0^{\circ}$.







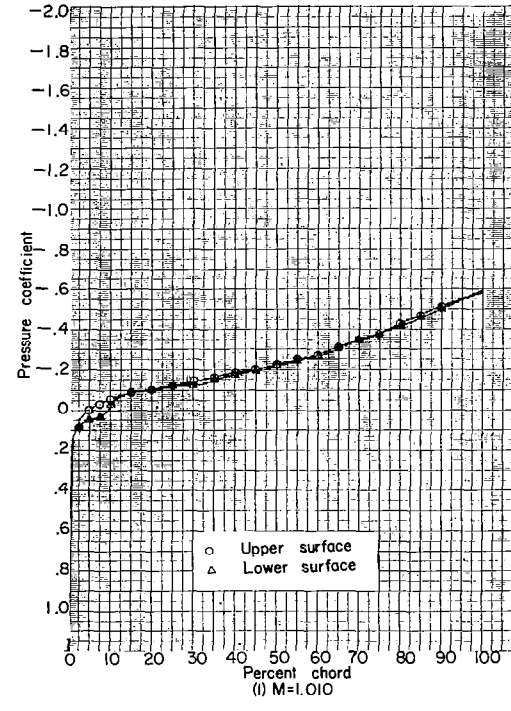
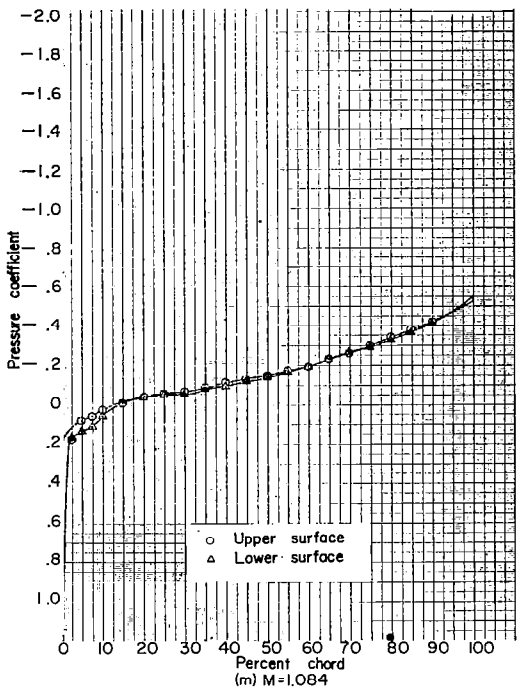
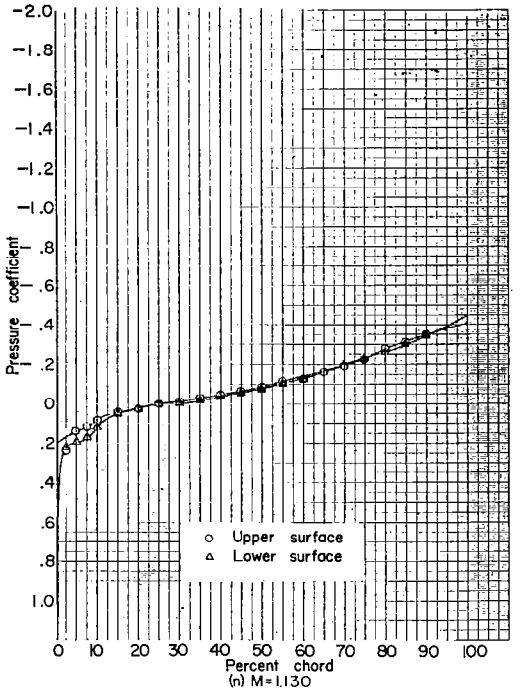
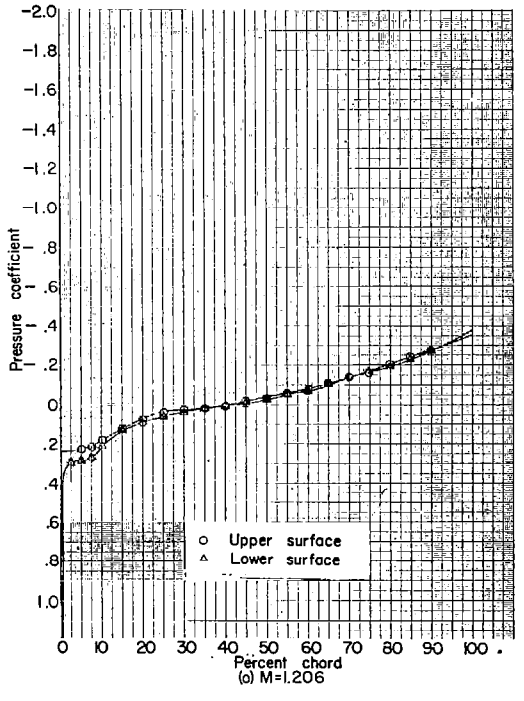


Figure 12.- Continued. NACA 16-006; $\alpha = 0^{\circ}$.







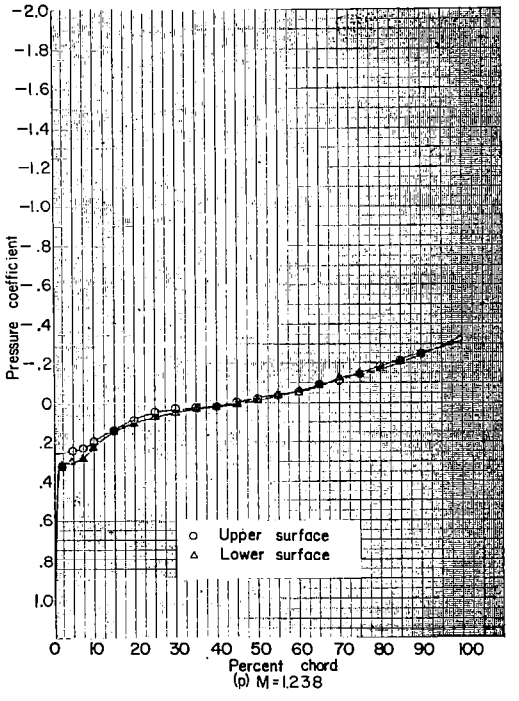


Figure 12.- Concluded. NACA 16-006; $\alpha = 0^{\circ}$.

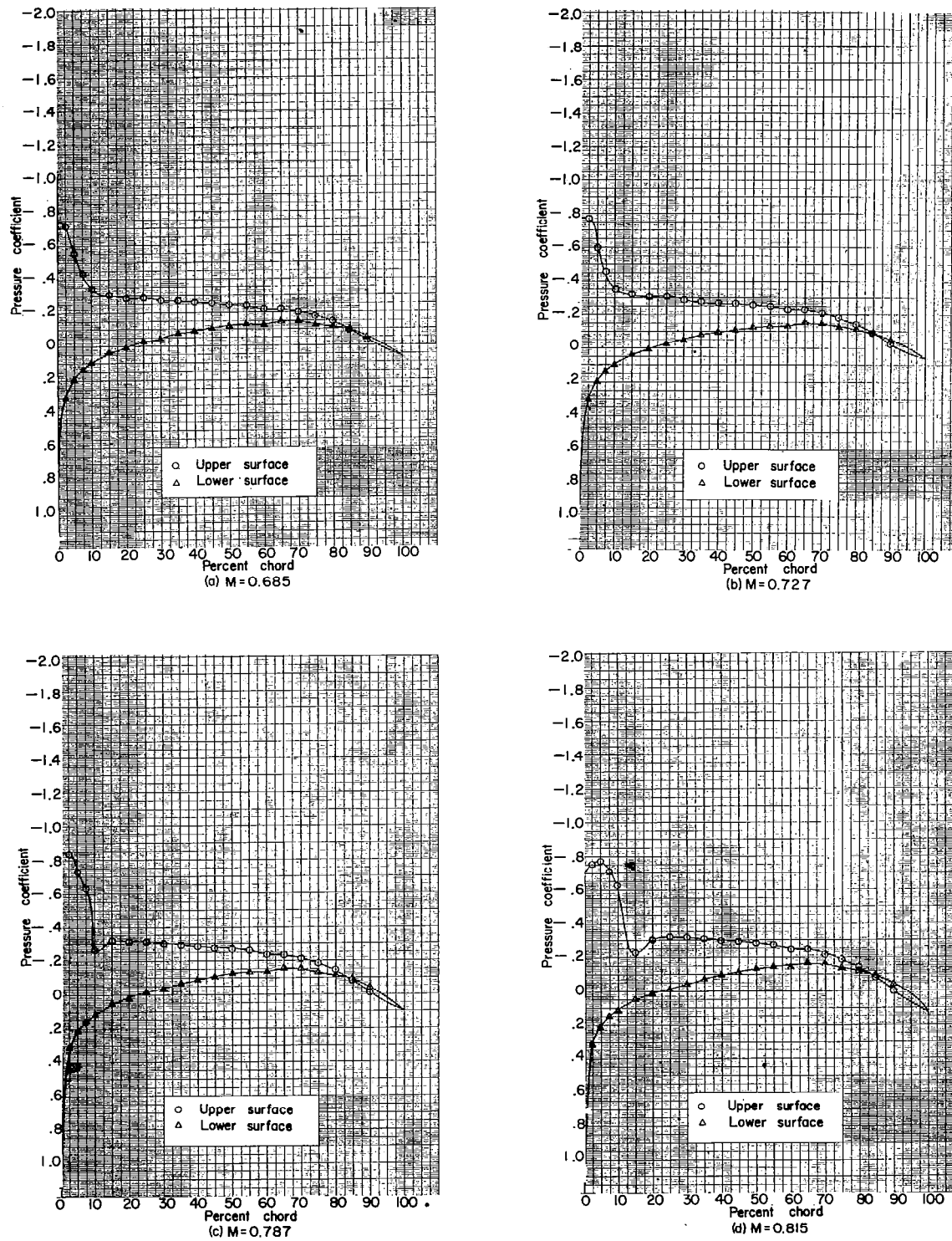
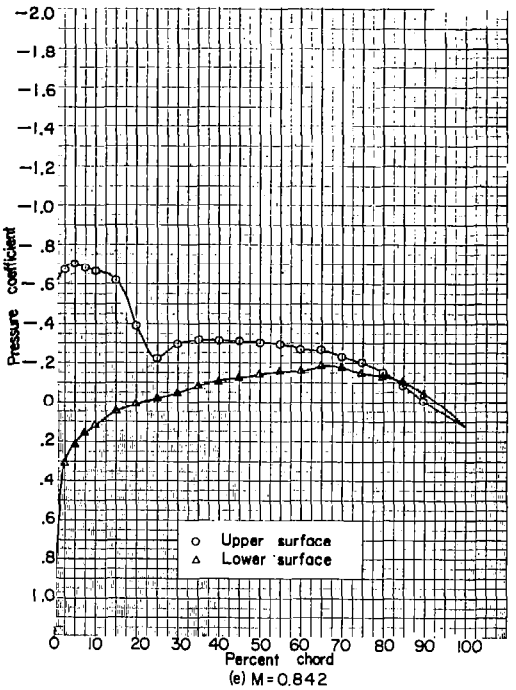
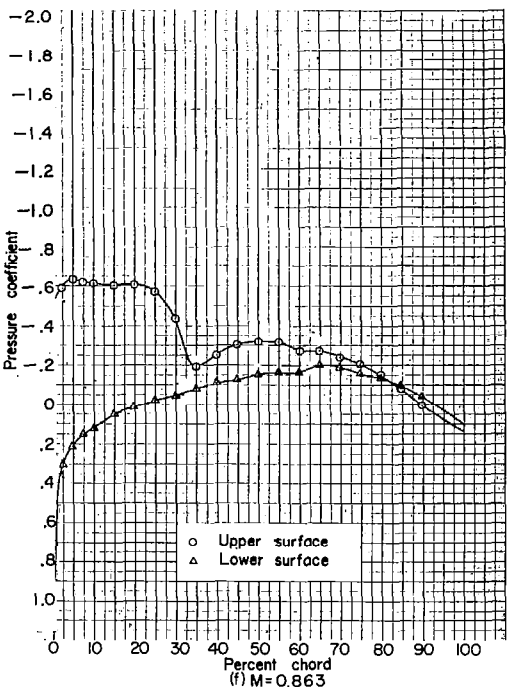
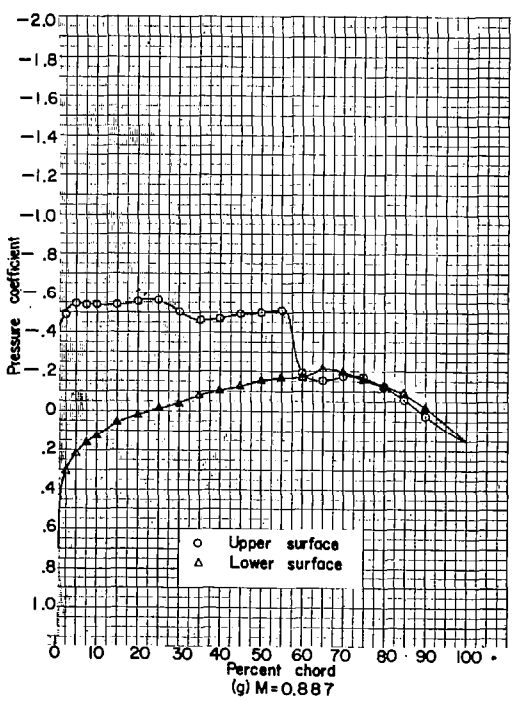


Figure 13.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 2^{\circ}$.







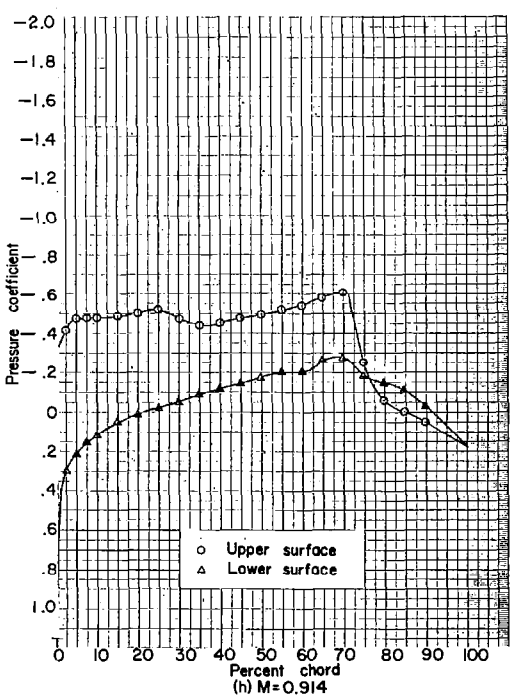
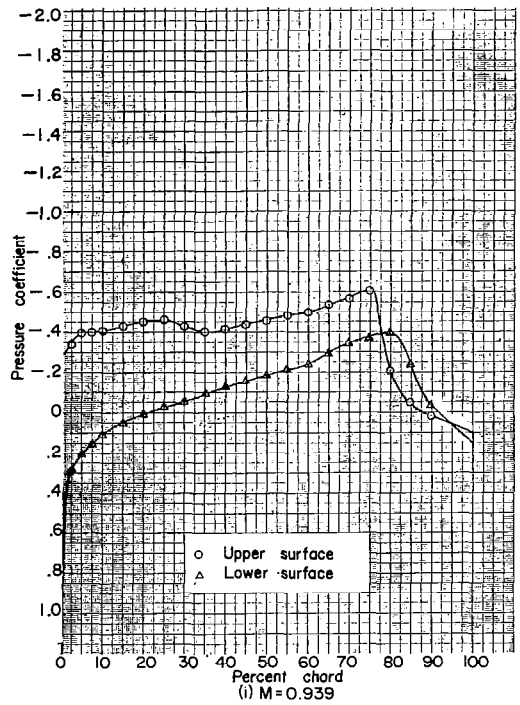
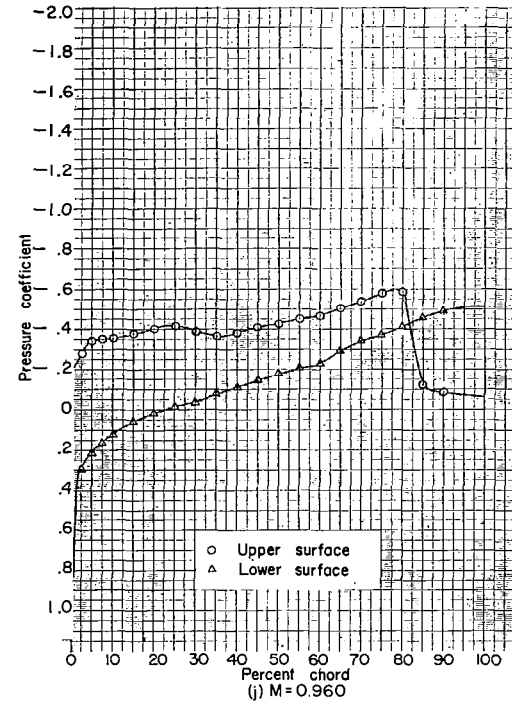
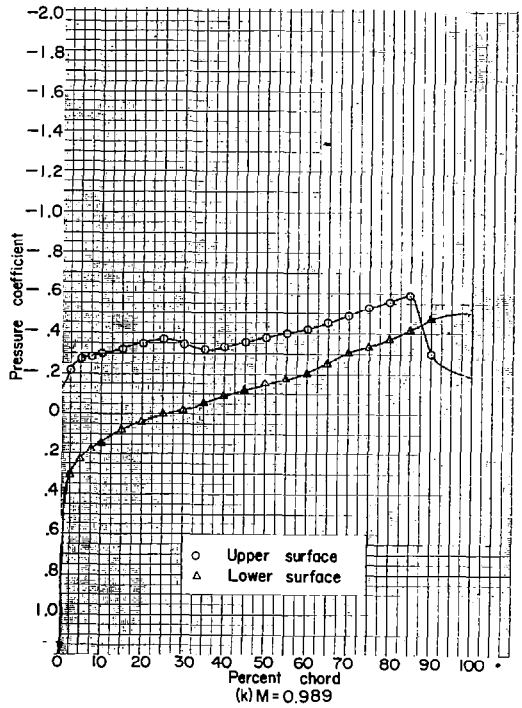


Figure 13.- Continued. NACA 16-006; $\alpha = 2^{\circ}$.







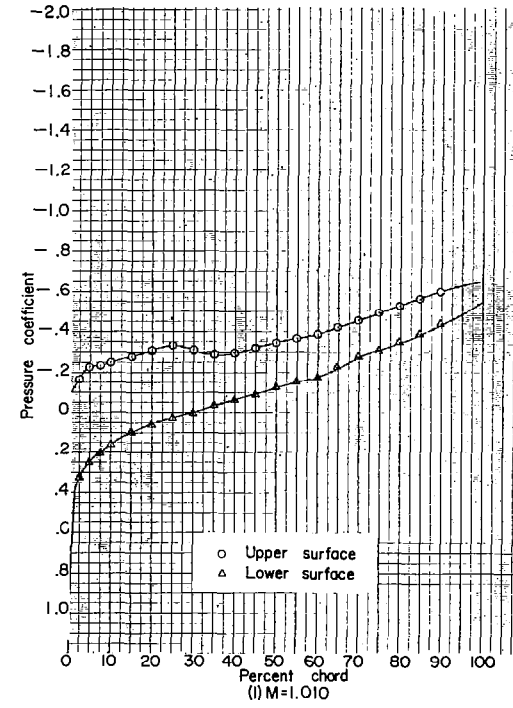
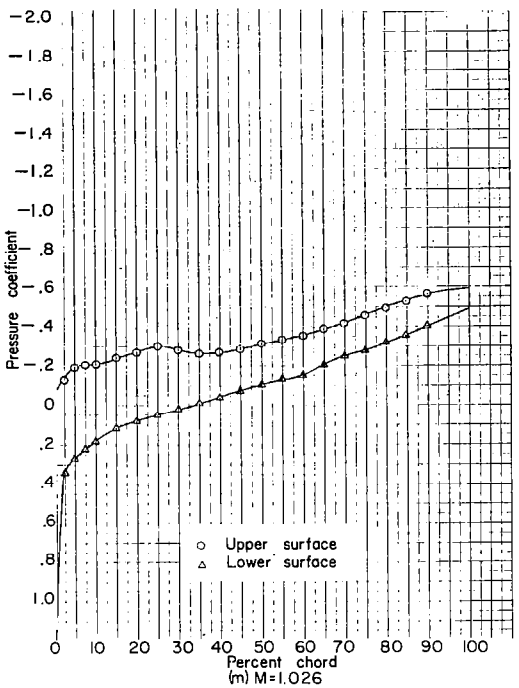
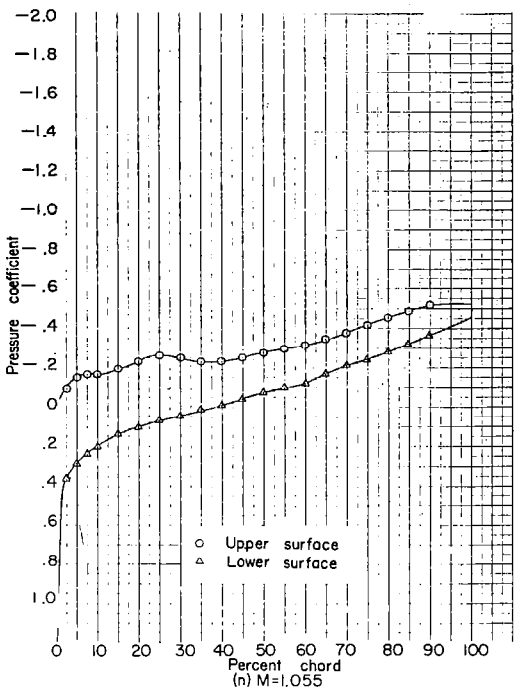
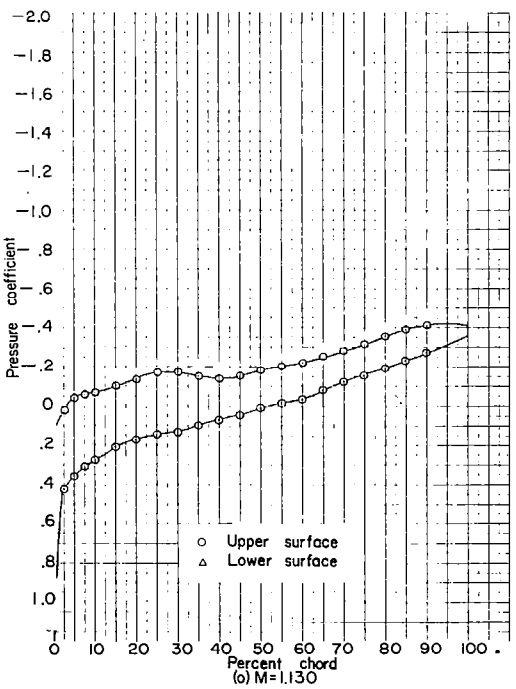


Figure 13.- Continued. NACA 16-006; $\alpha = 2^{\circ}$.







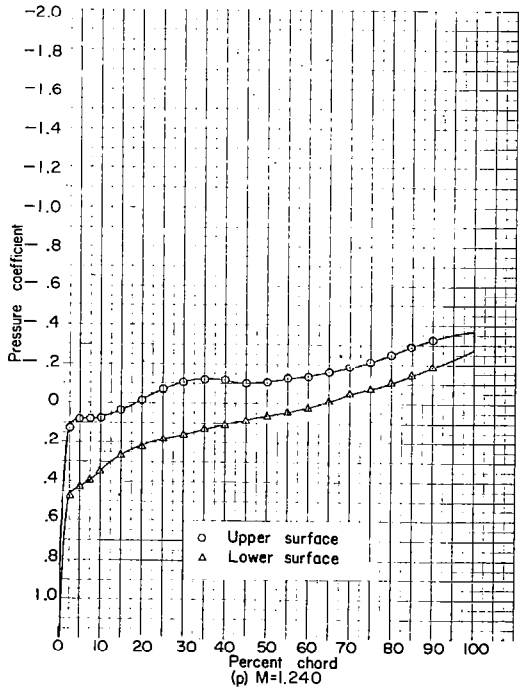


Figure 13.- Concluded. NACA 16-006; $\alpha = 2^{\circ}$.

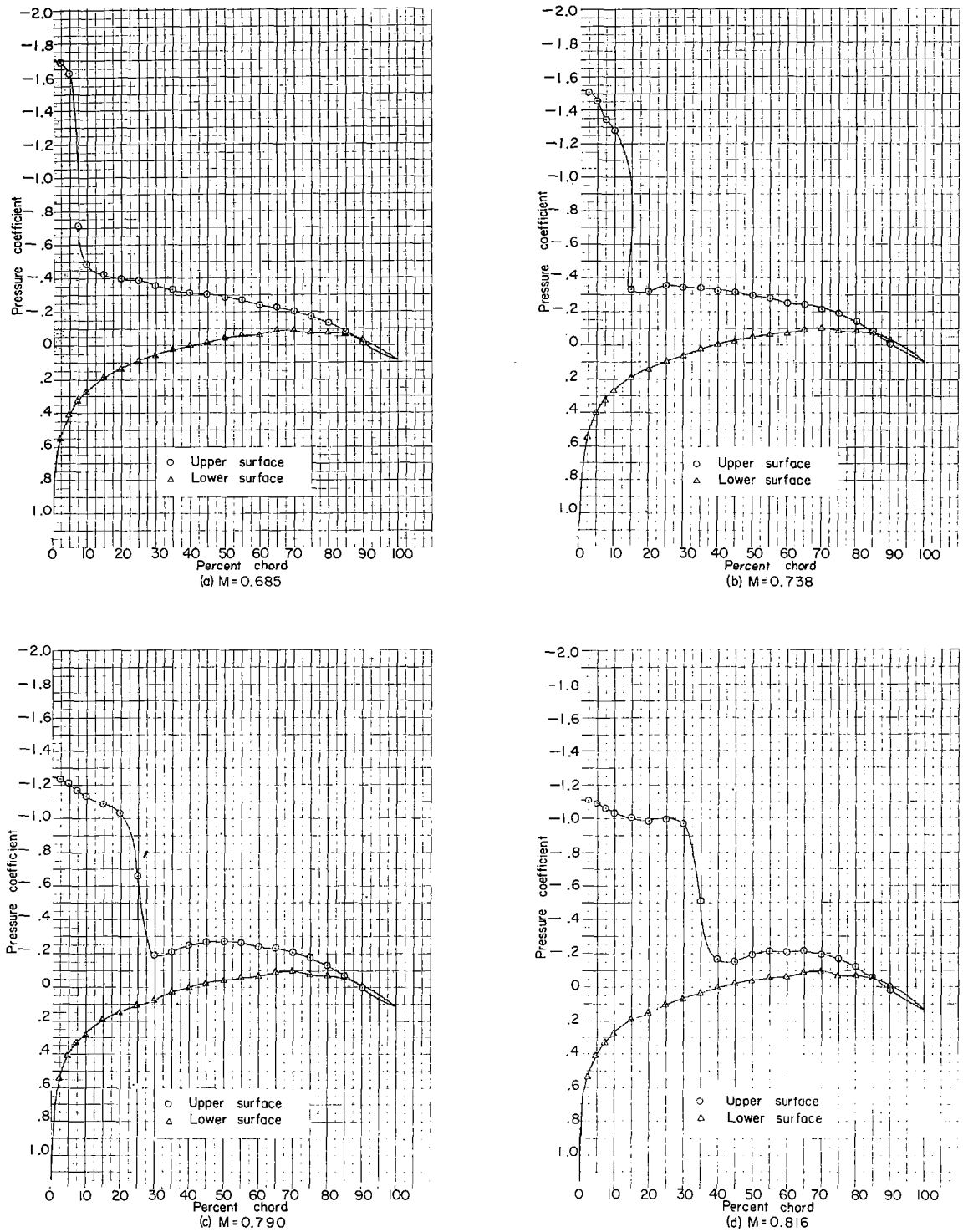
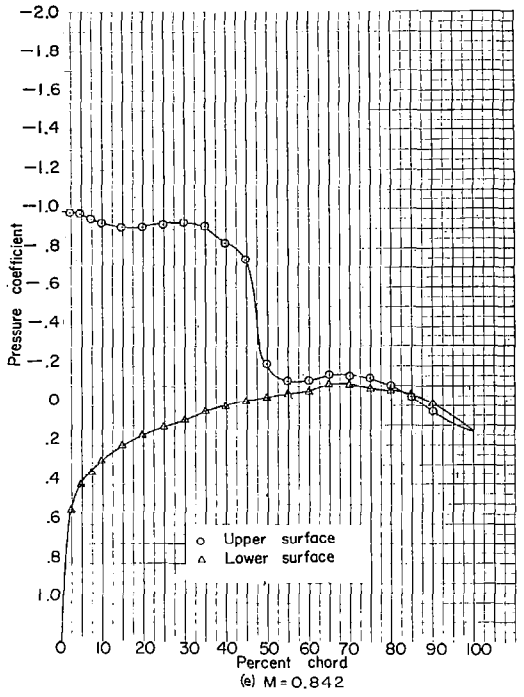
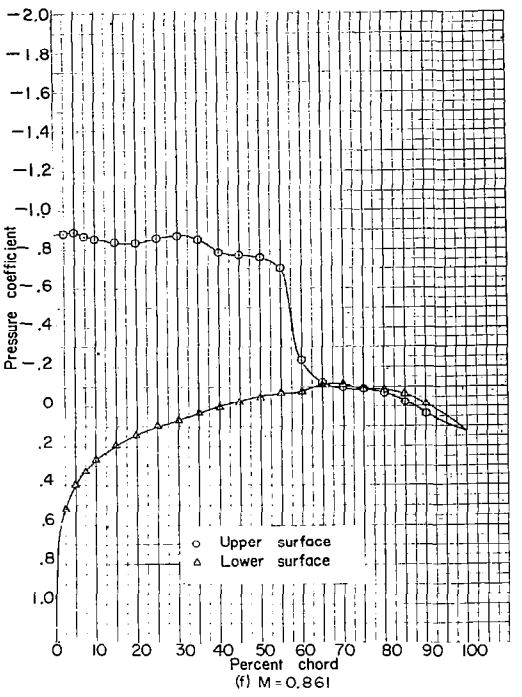
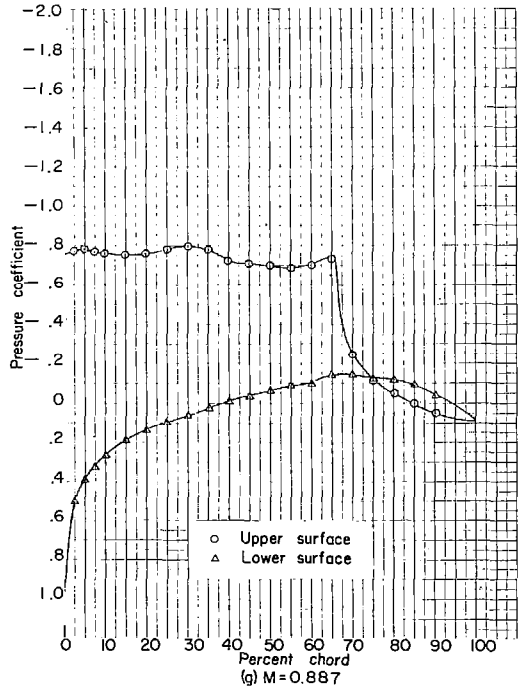


Figure 14.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 4^{o}$.







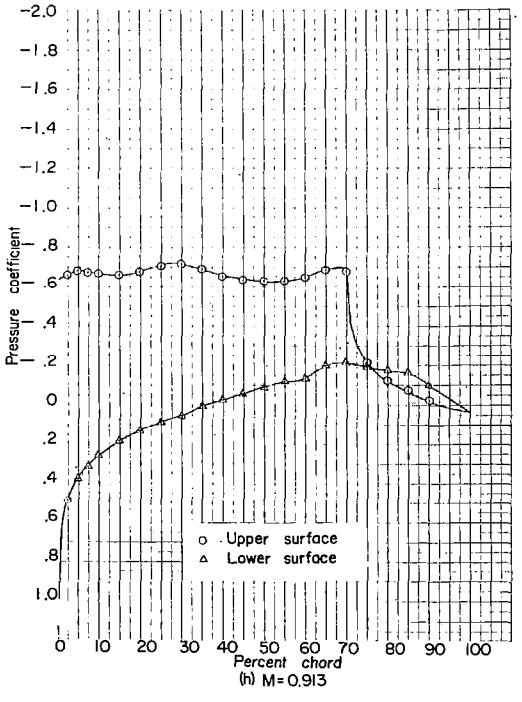
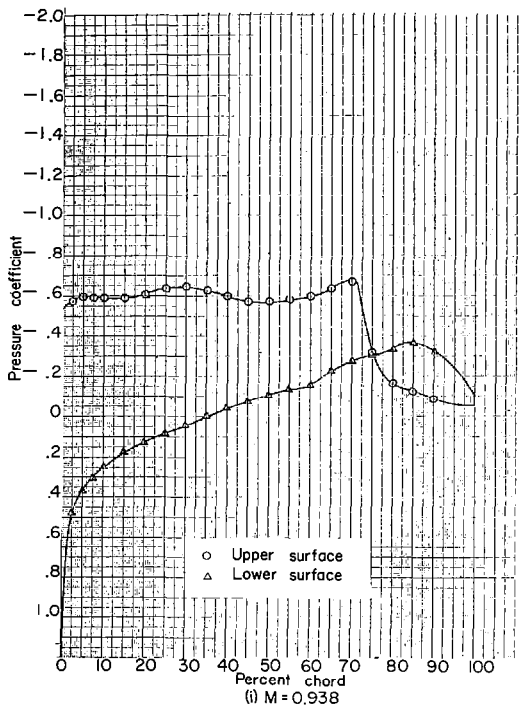
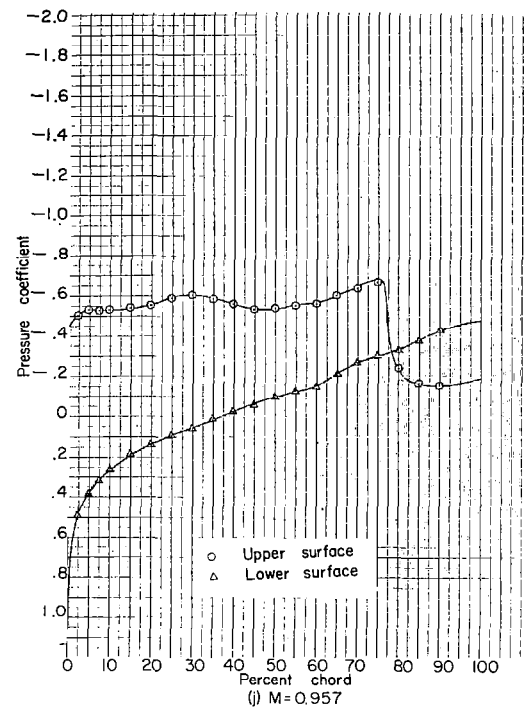


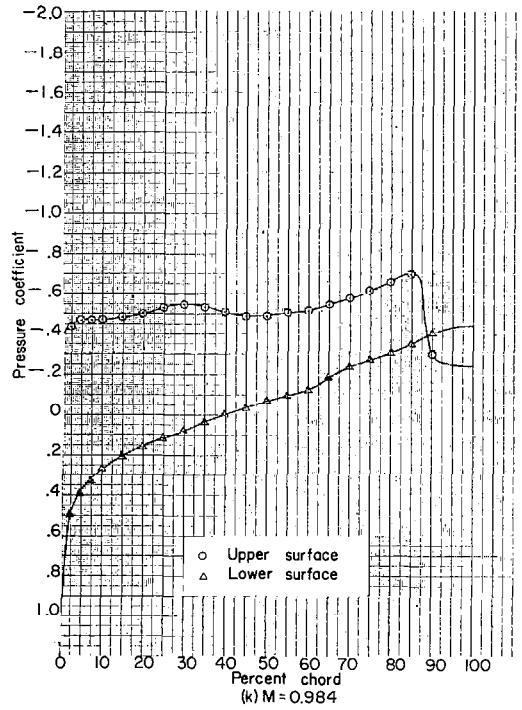
Figure 14.- Continued. NACA 16-006; $\alpha = 4^{\circ}$.



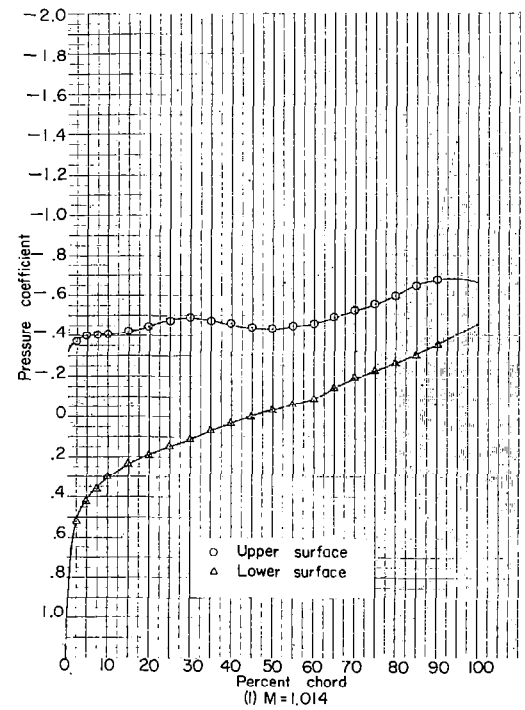
(i) M = 0.938



(j) M = 0.957

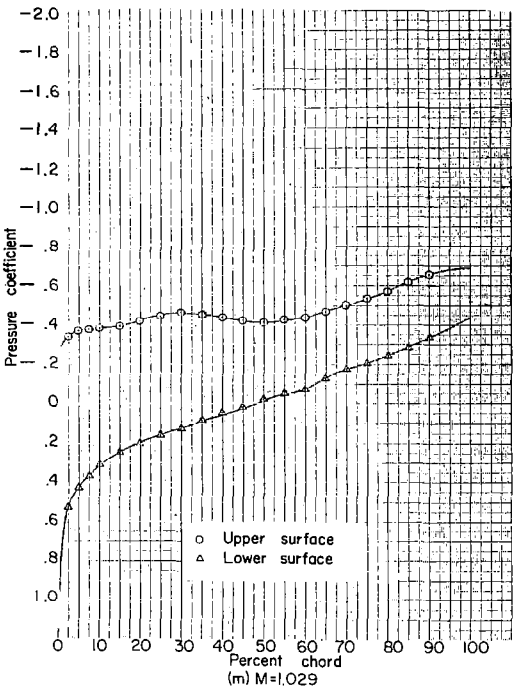


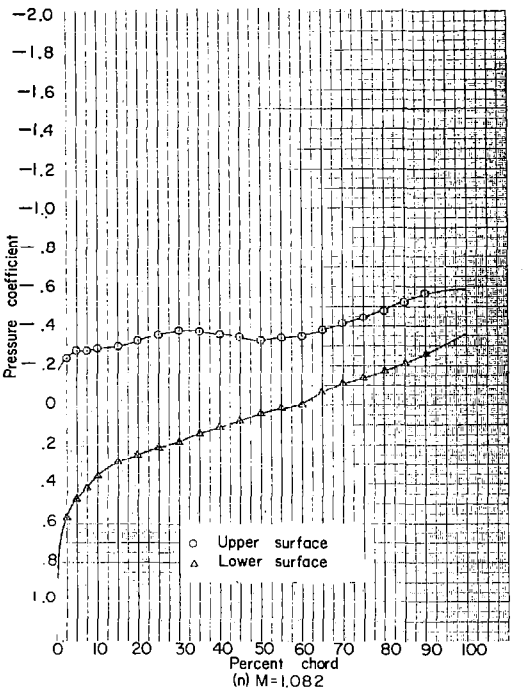
(k) M = 0.984

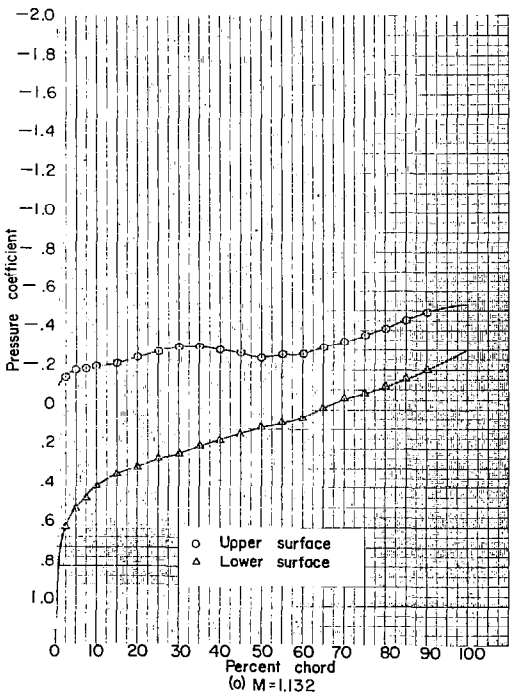


(l) M = 1.014

Figure 14.- Continued. NACA 16-006; $\alpha = 4^{\circ}$.







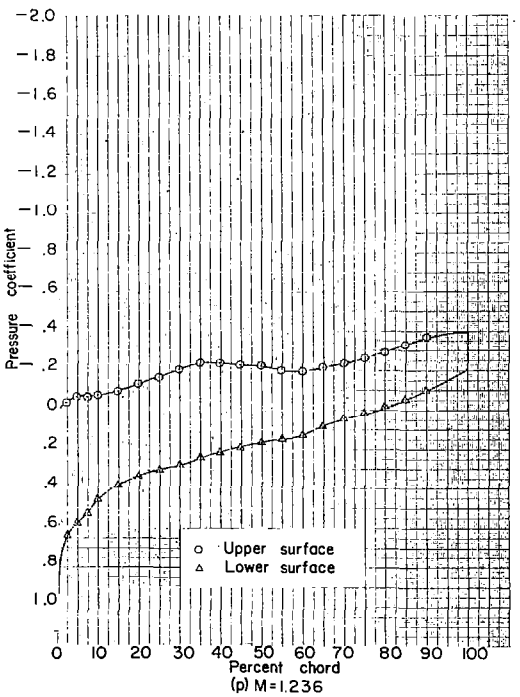


Figure 14.- Concluded. NACA 16-006; $\alpha = 4^{\circ}$.

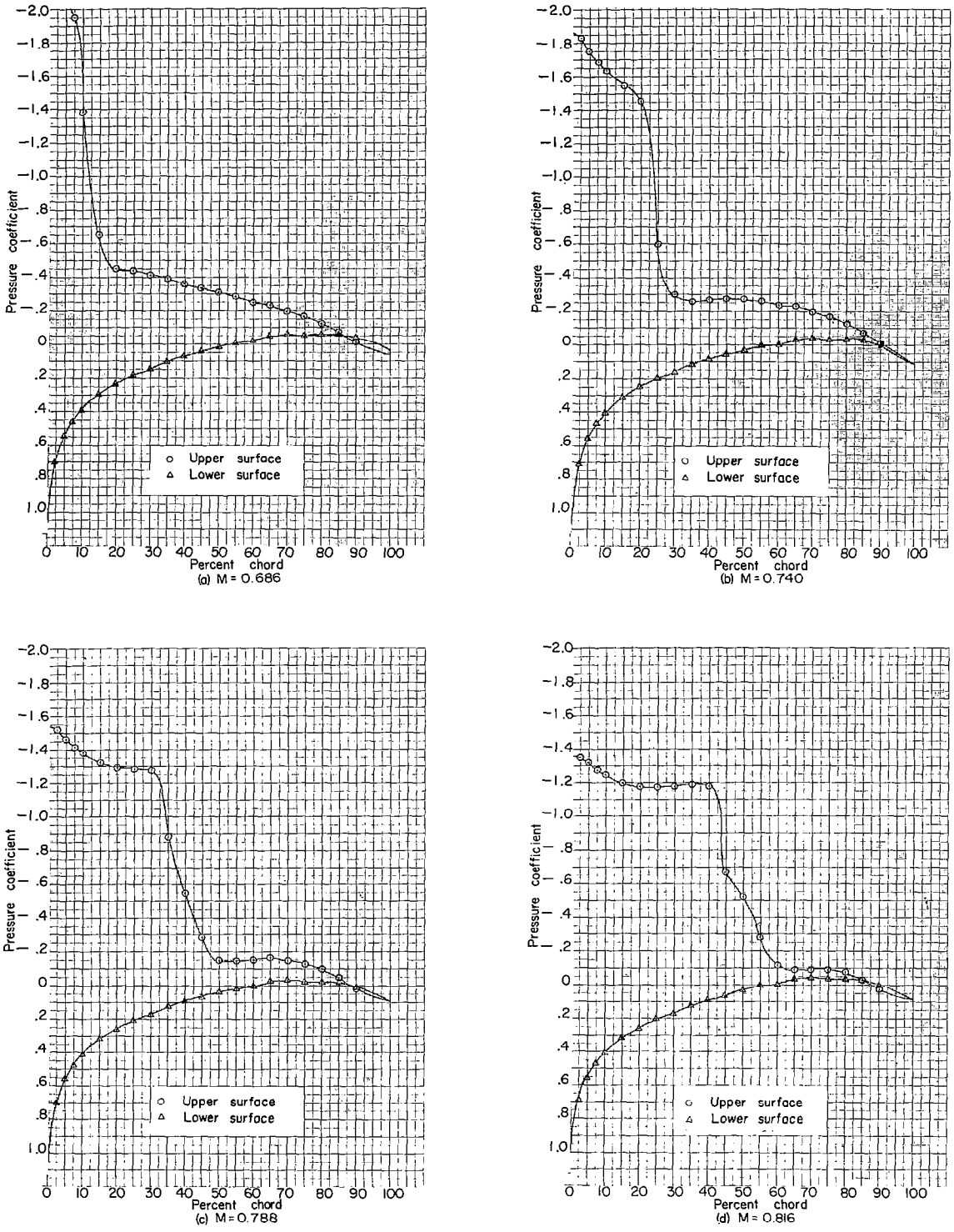
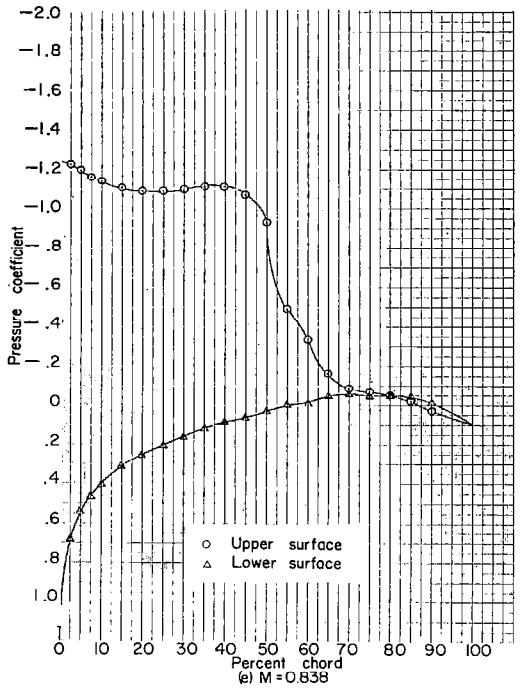
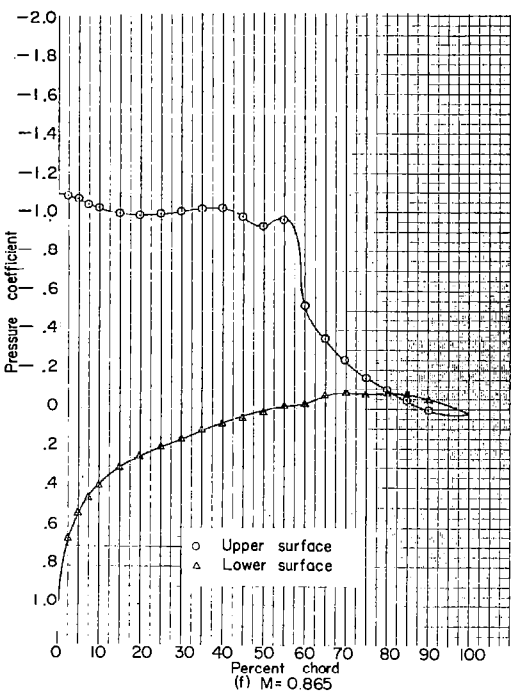
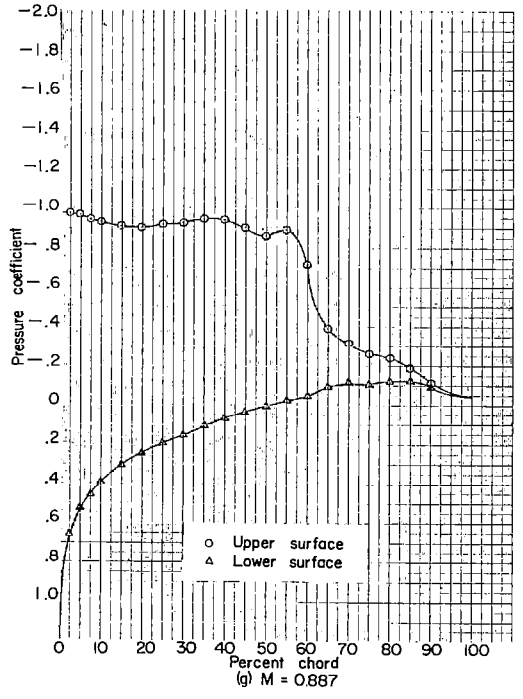


Figure 15.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 6^{\circ}$.







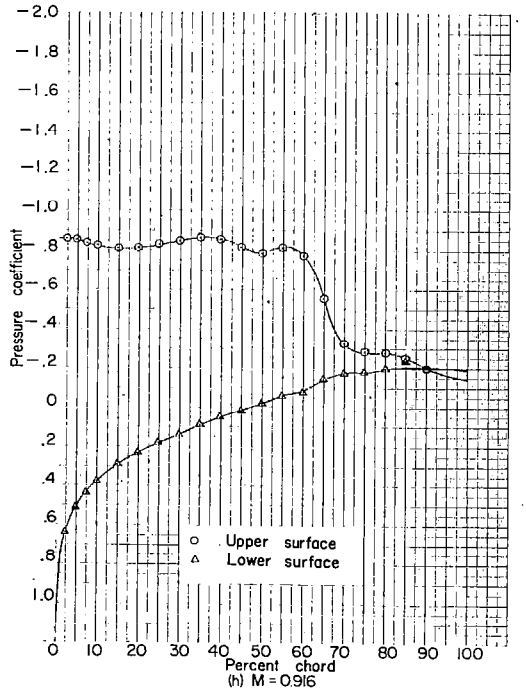


Figure 15.- Continued. NACA 16-006; $\alpha = 6^{\circ}$.

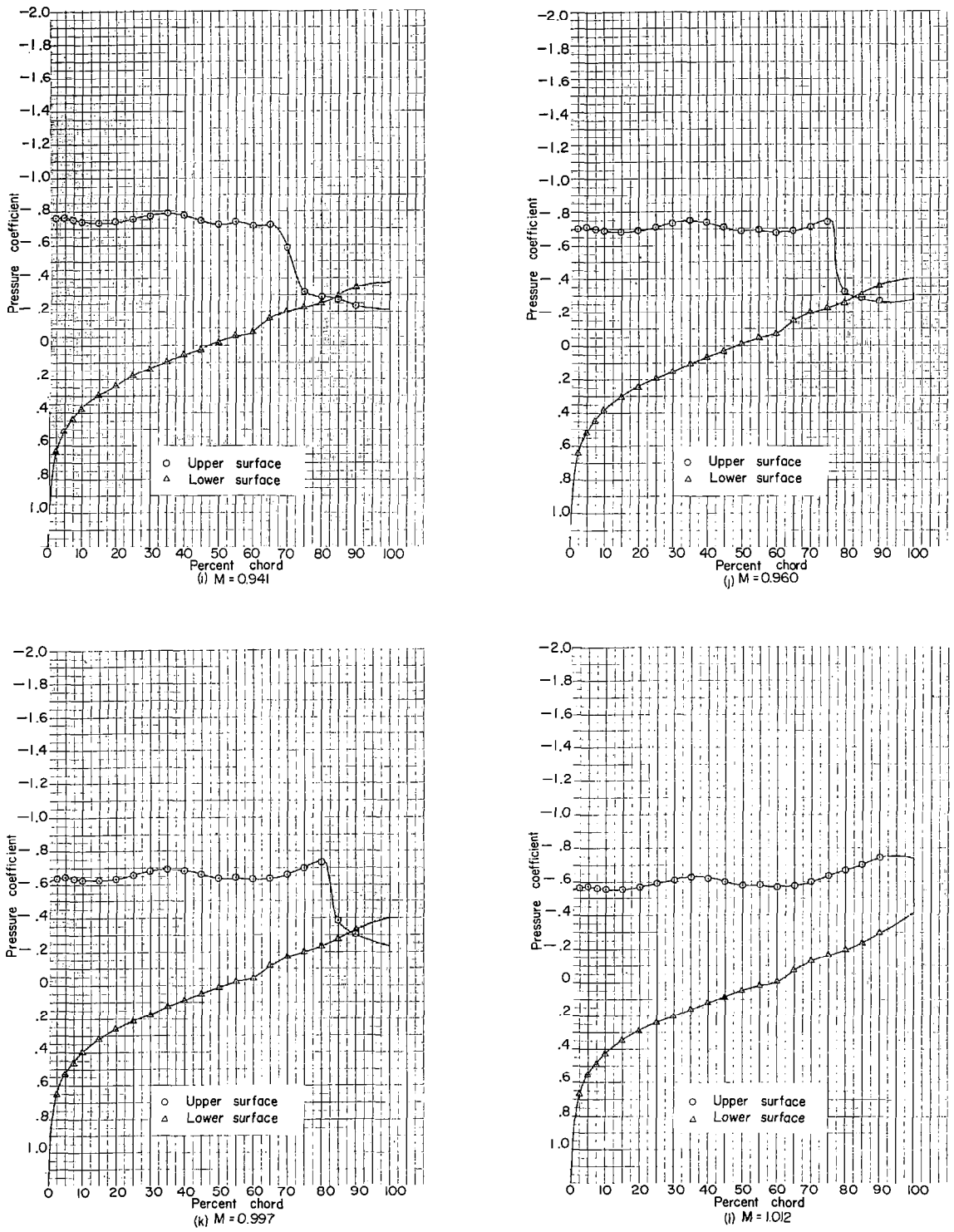
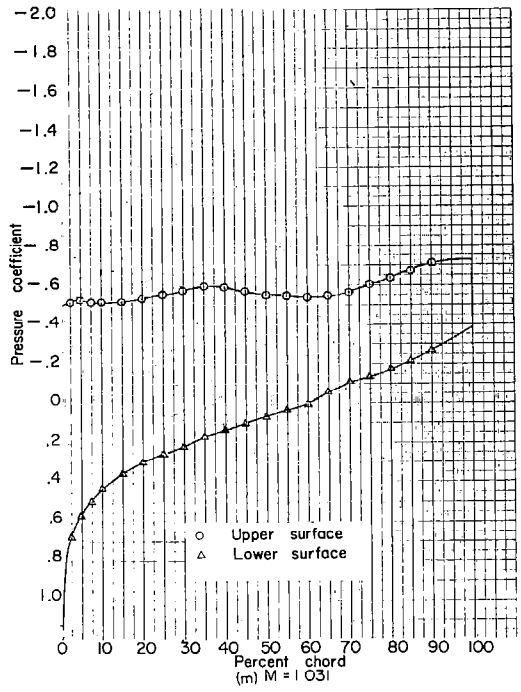
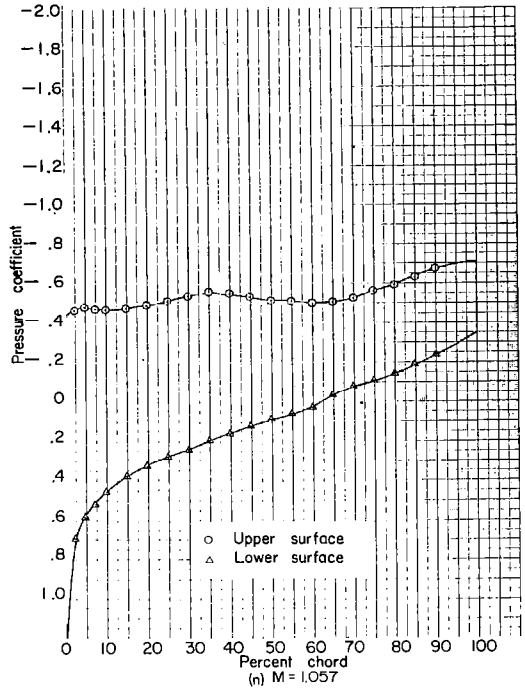
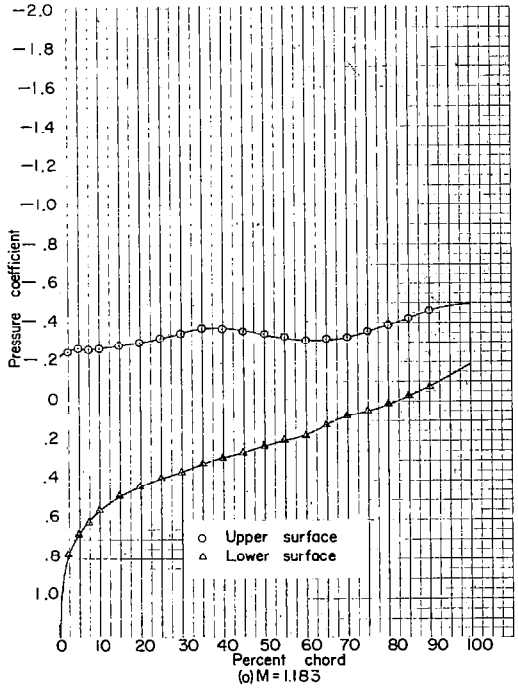


Figure 15.- Continued. NACA 16-006; $\alpha = 6^{\circ}$.







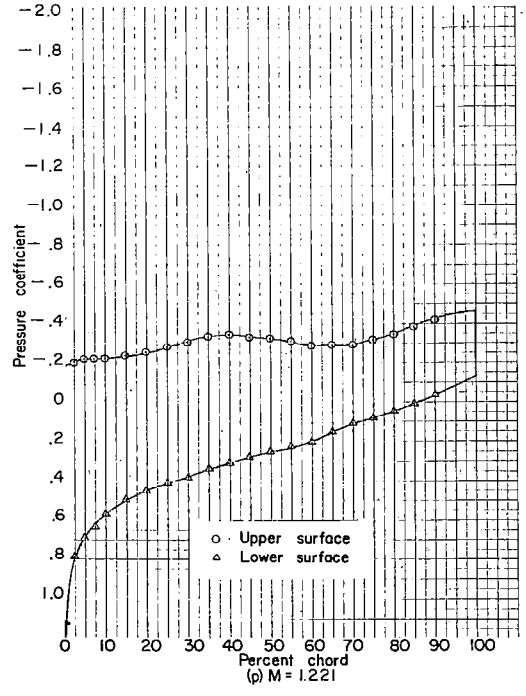


Figure 15.- Concluded. NACA 16-006; $\alpha = 6^{\circ}$.

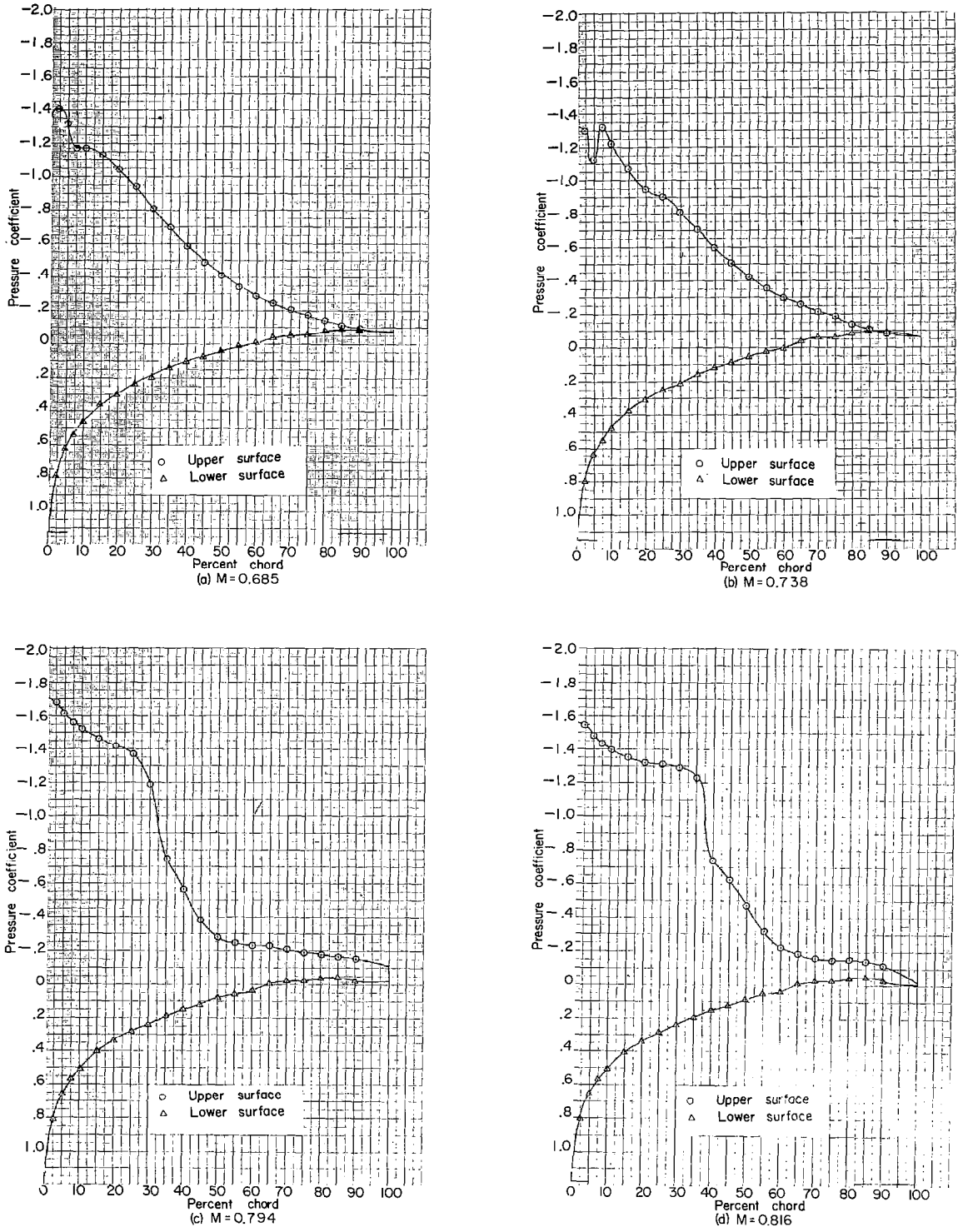
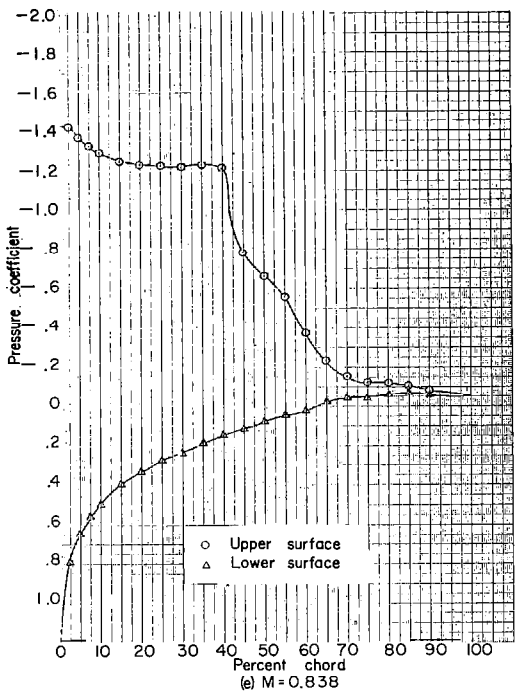
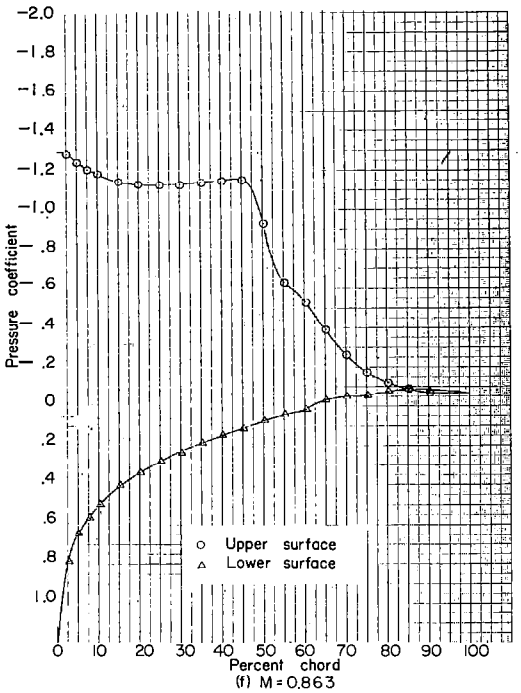
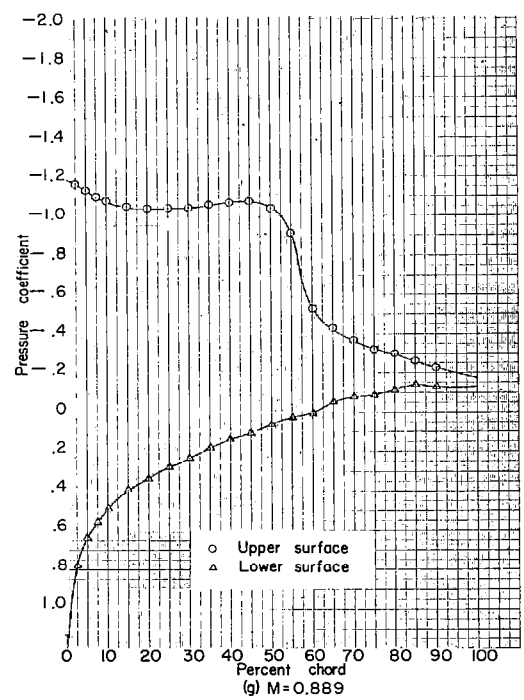


Figure 16.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 8^{o}$.







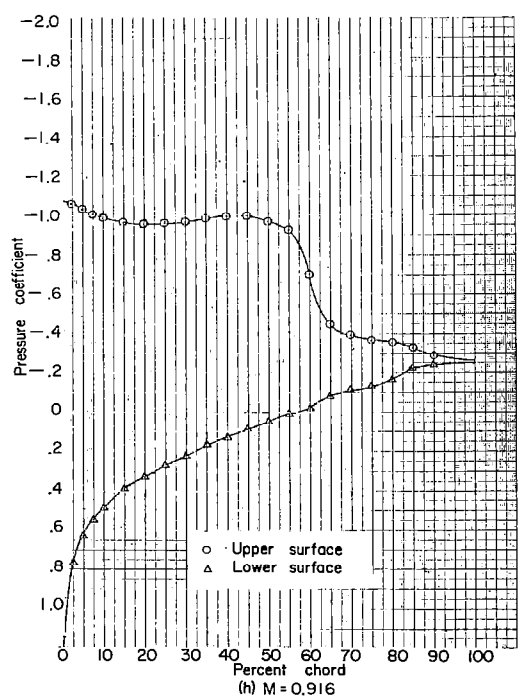
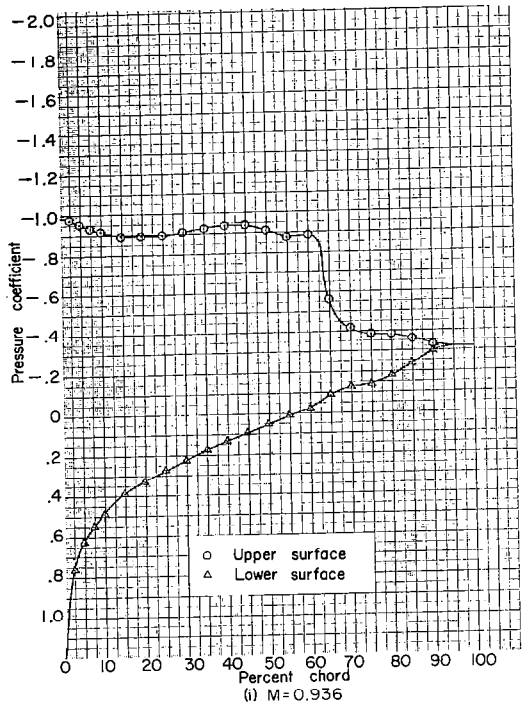
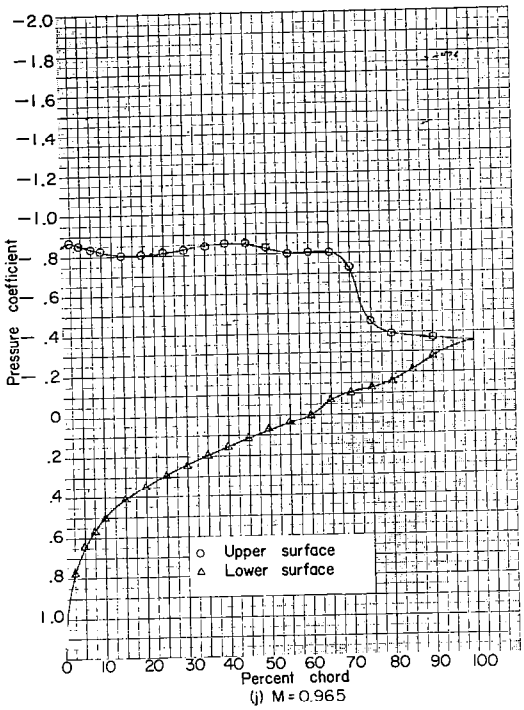


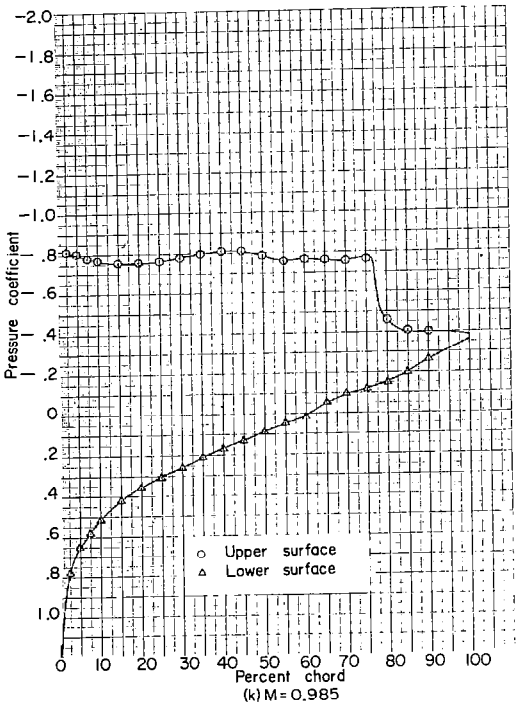
Figure 16.- Continued. NACA 16-006; $\alpha = 8^{\circ}$.



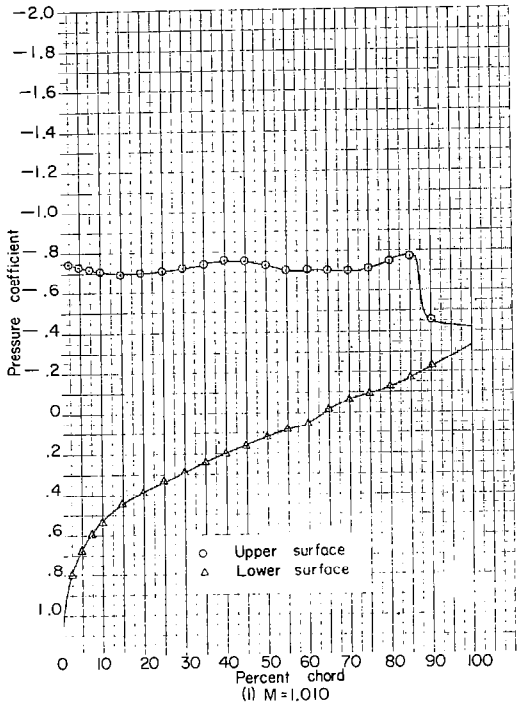
(i) M = 0.936



(j) M = 0.965

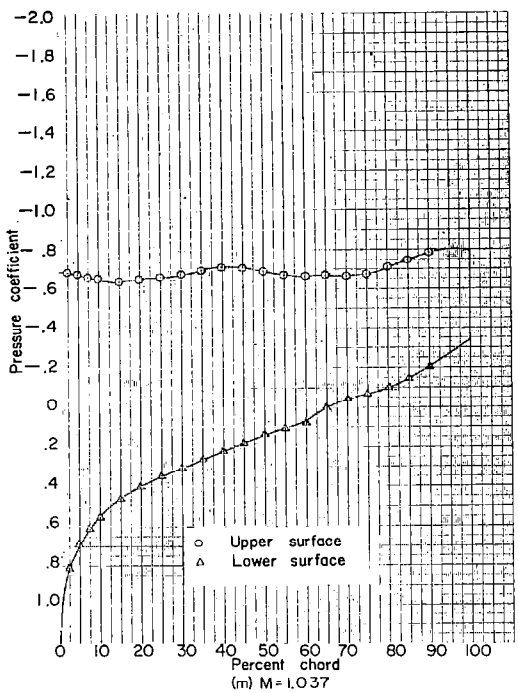


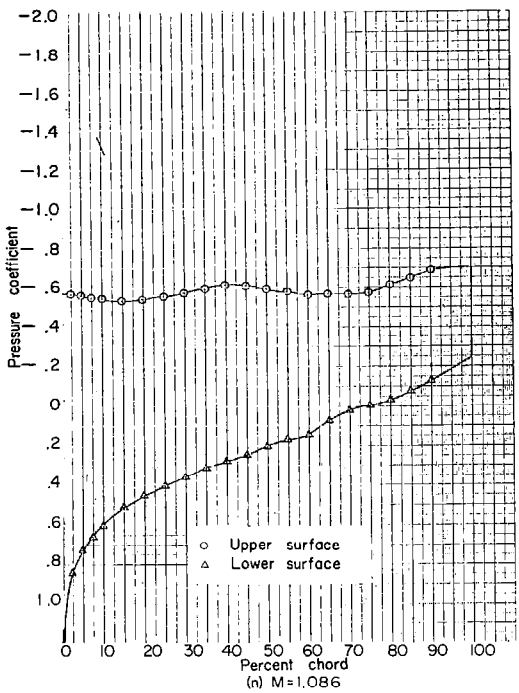
(k) M = 0.985

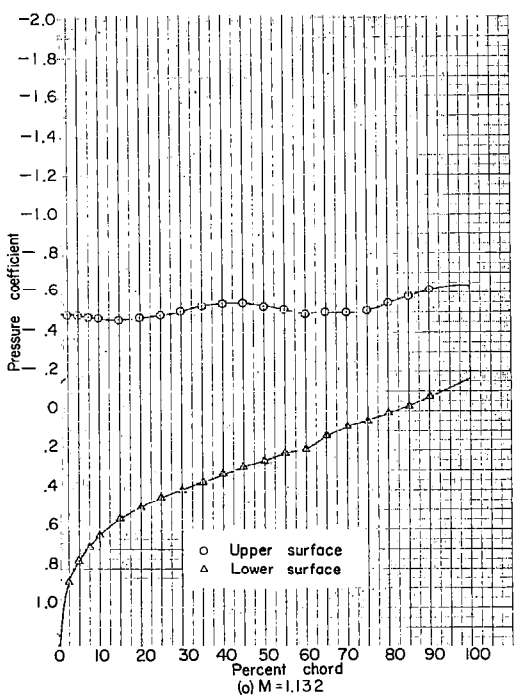


(l) M = 1.010

Figure 16.- Continued. NACA 16-006; $\alpha = 8^{\circ}$.







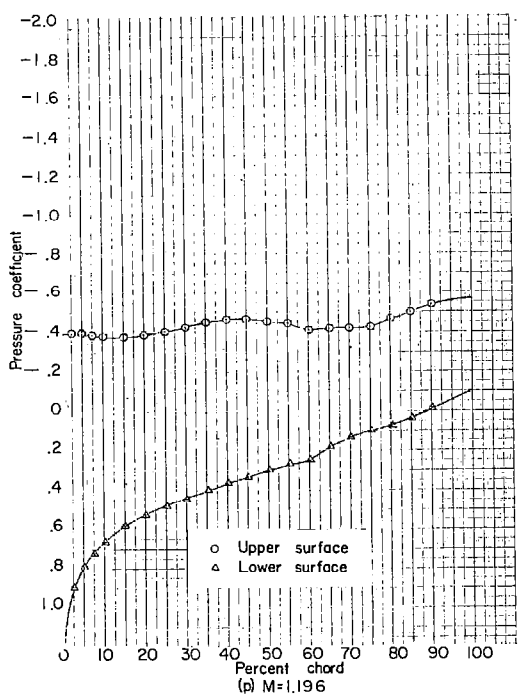


Figure 16.- Concluded. NACA 16-006; $\alpha = 8^{\circ}$.

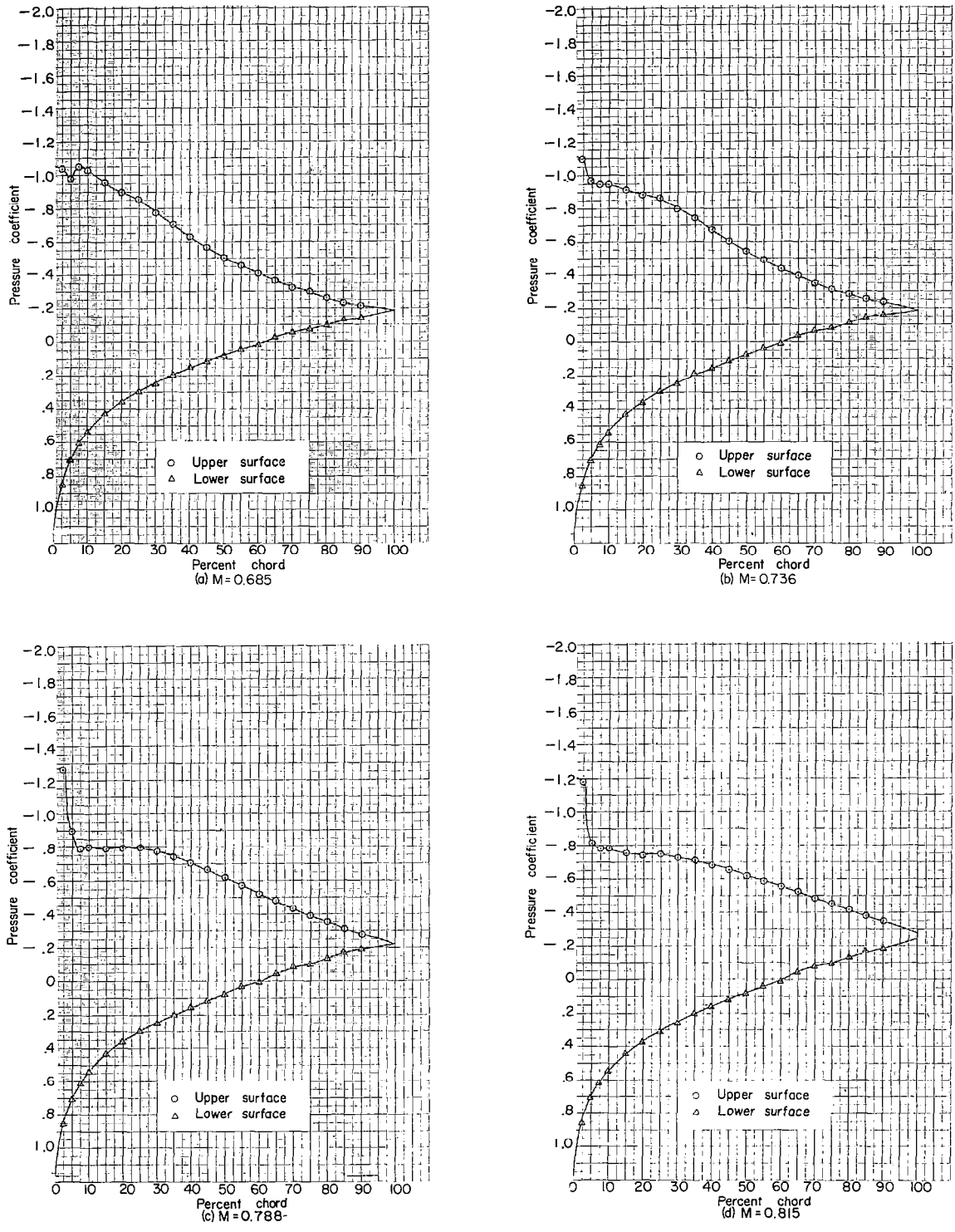
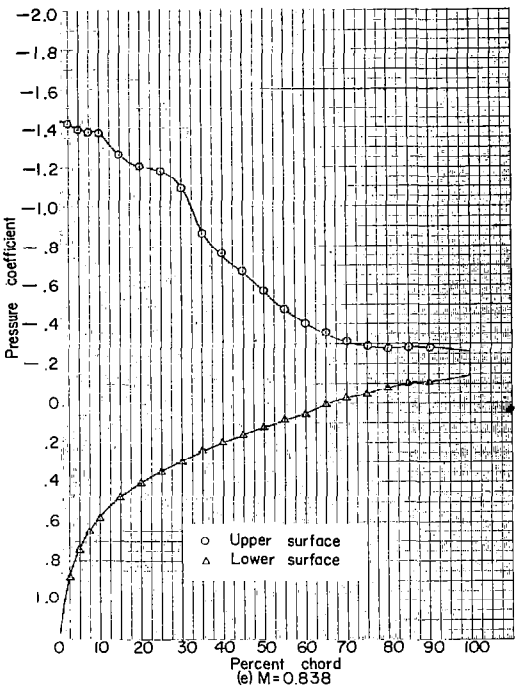
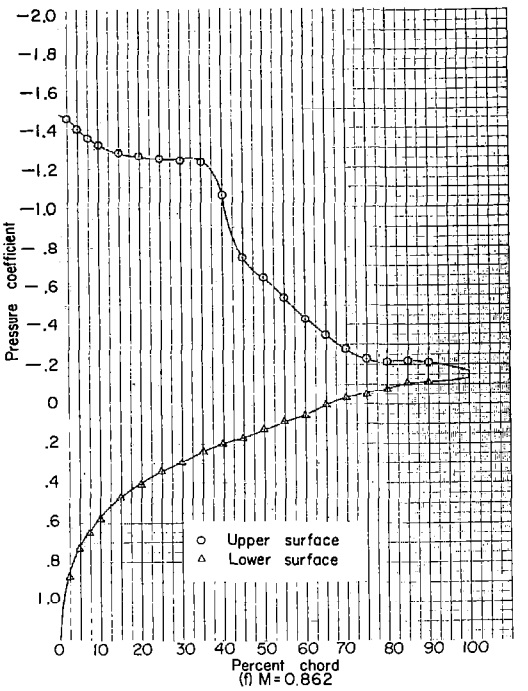
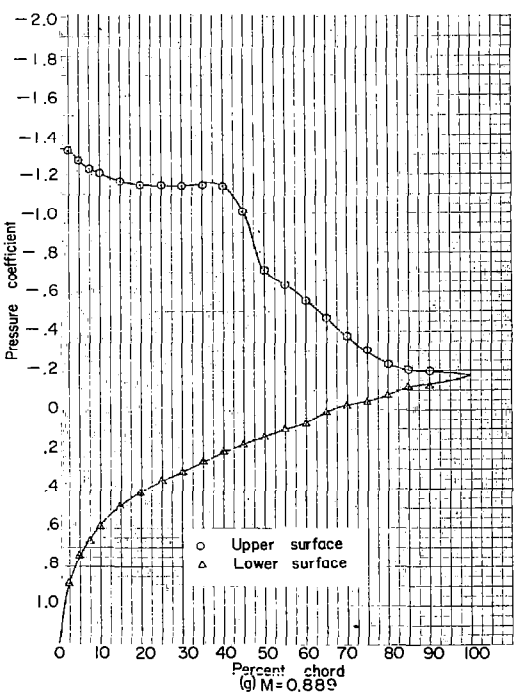


Figure 17.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 10^{\circ}$.







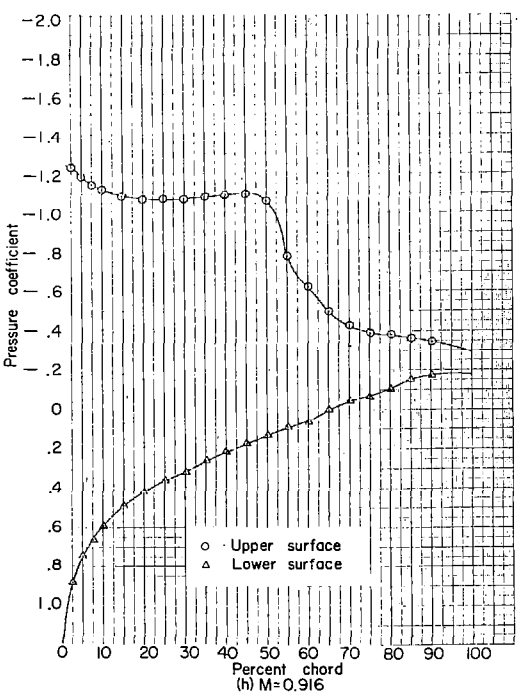
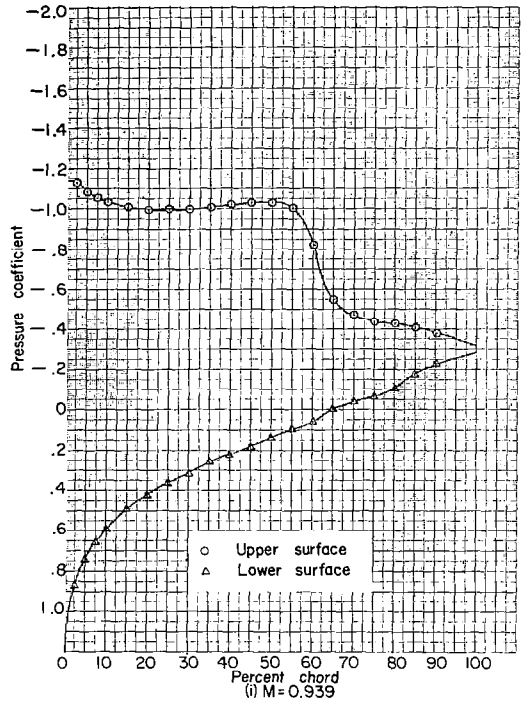
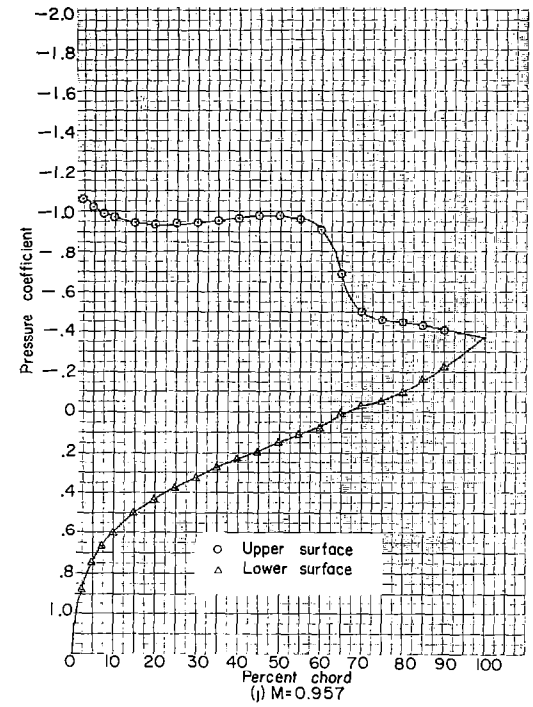
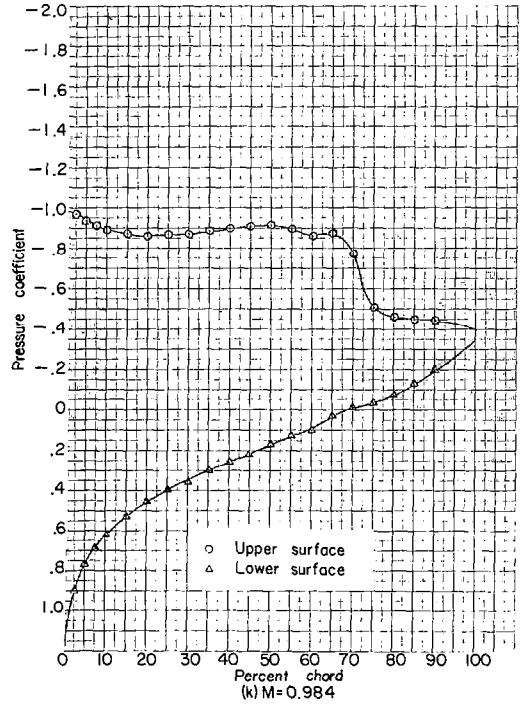


Figure 17.- Continued. NACA 16-006; $\alpha = 10^{\circ}$.







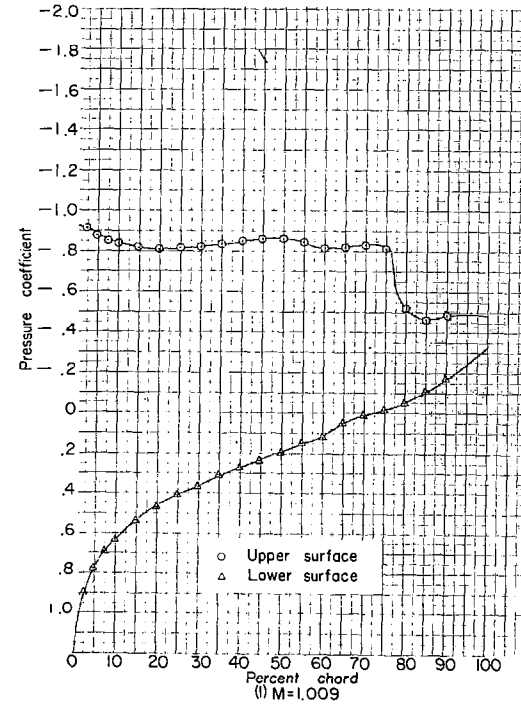
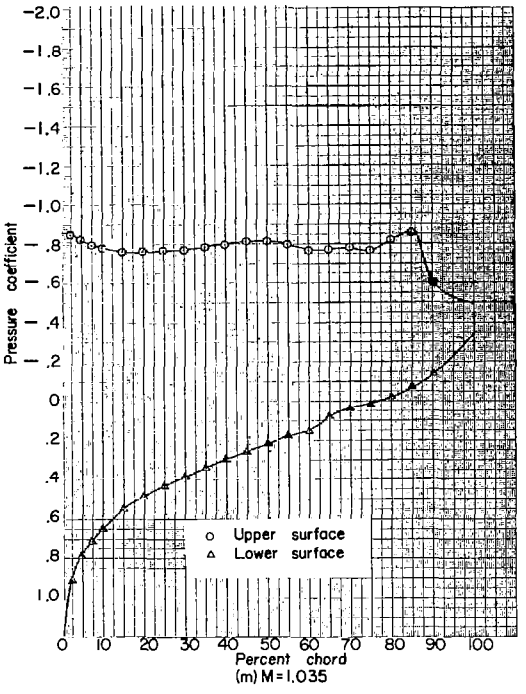
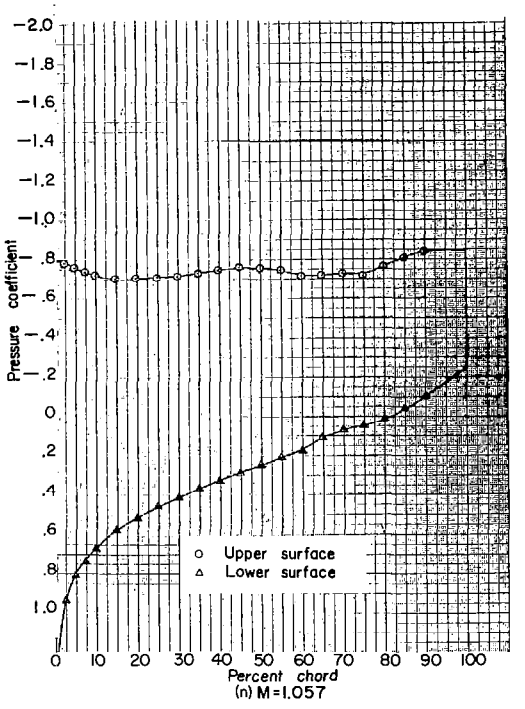
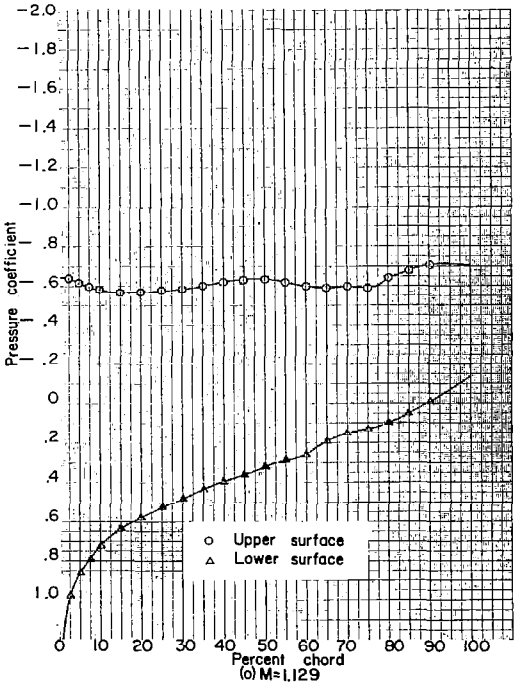


Figure 17.- Continued. NACA 16-006; $\alpha = 10^{\circ}$.







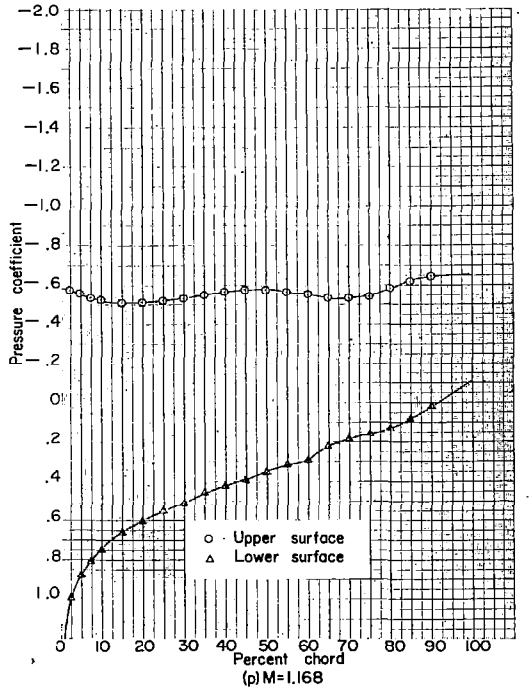


Figure 17.- Concluded. NACA 16-006; $\alpha = 10^{\circ}$.

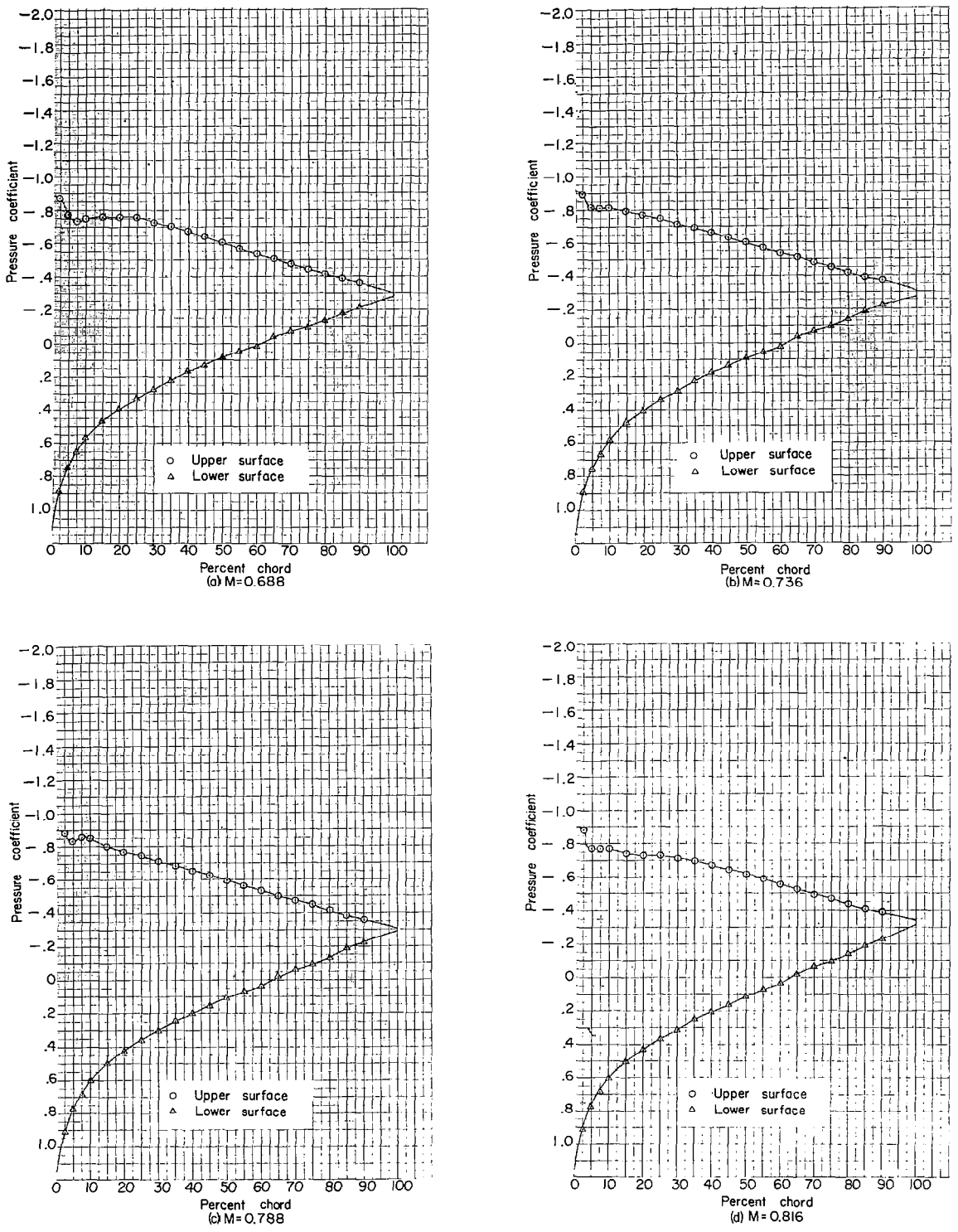
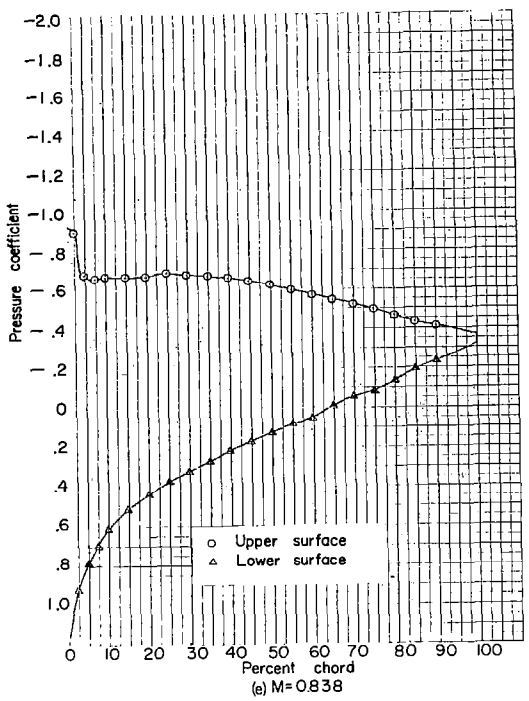
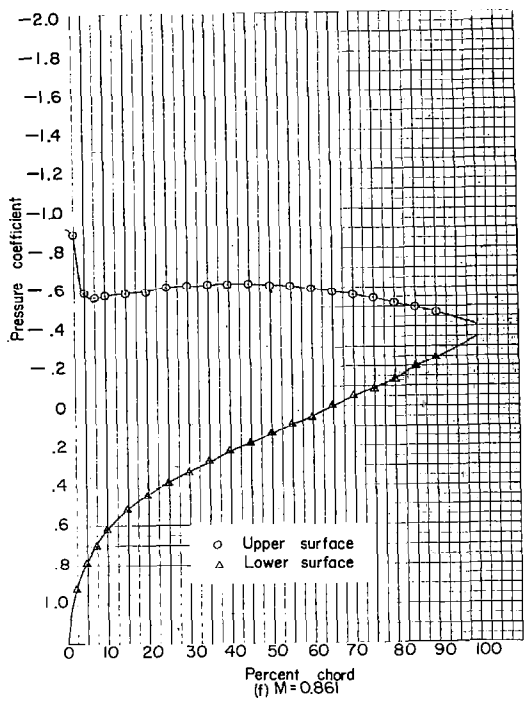
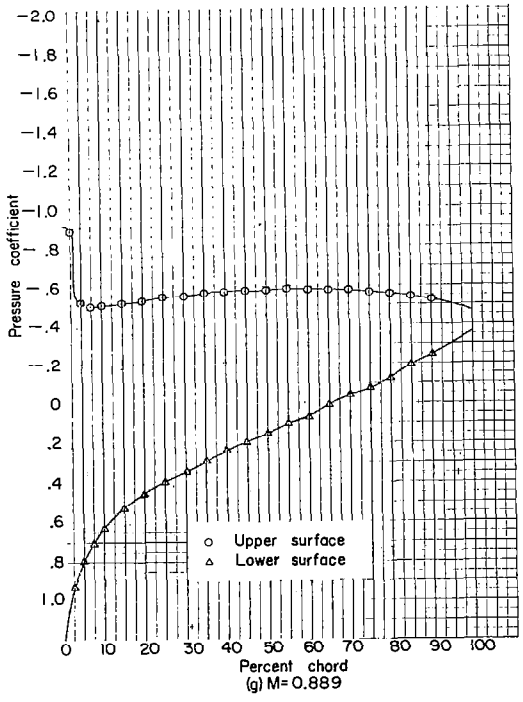


Figure 18.- Pressure distributions over NACA 16-006 airfoil section. $\alpha = 12^{\circ}$.







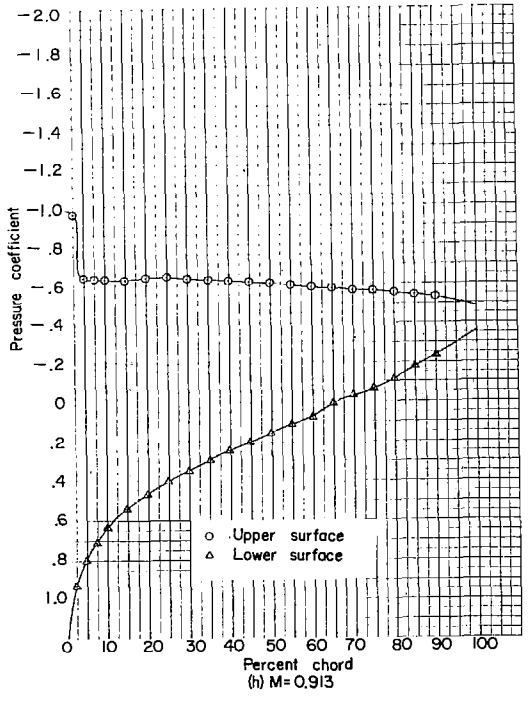
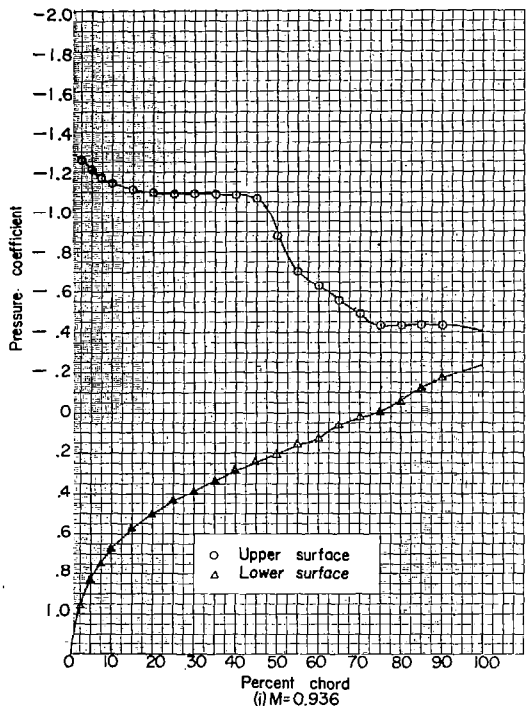
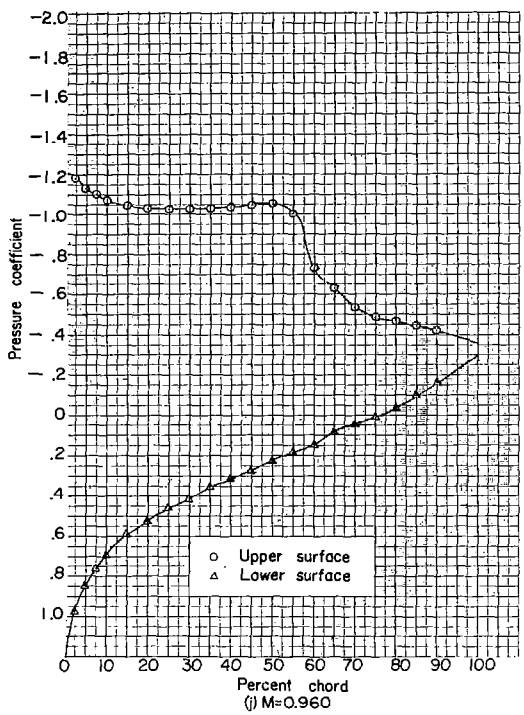


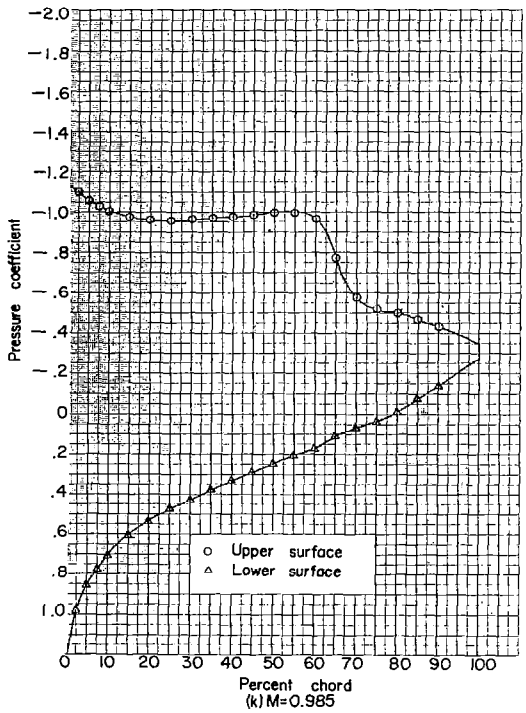
Figure 18.- Continued. NACA 16-006; $\alpha = 12^{\circ}$.



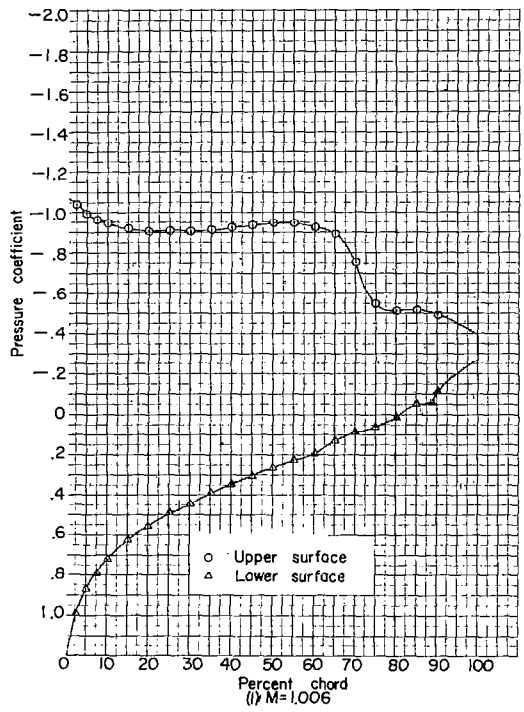
(i) M=0.936



(j) M=0.960

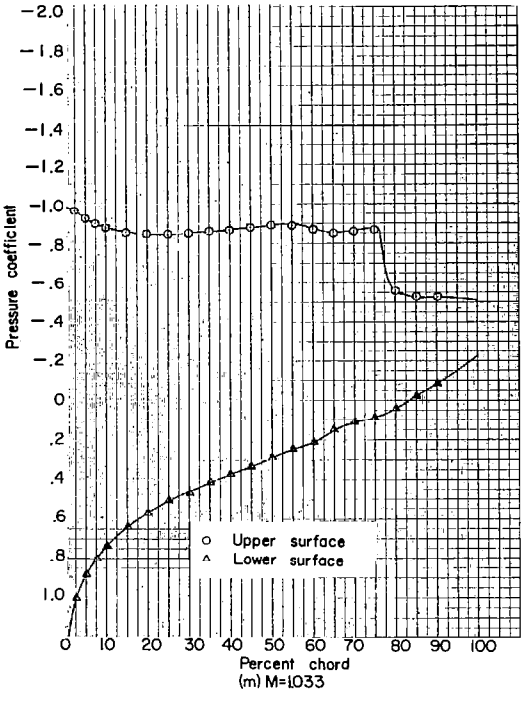


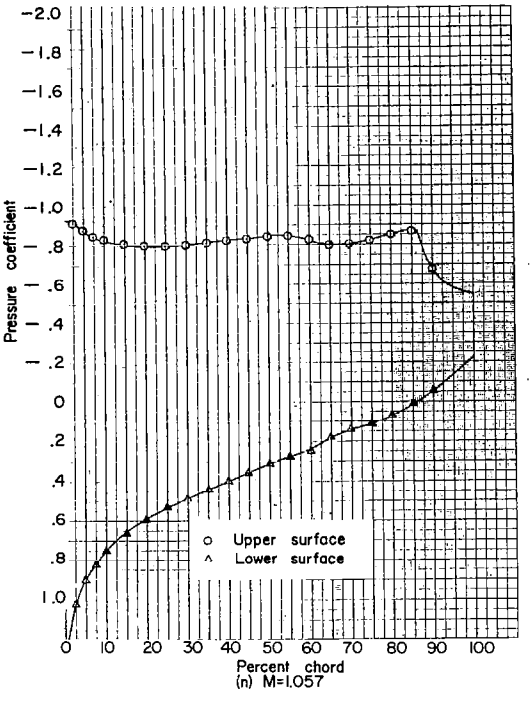
(k) M=0.985

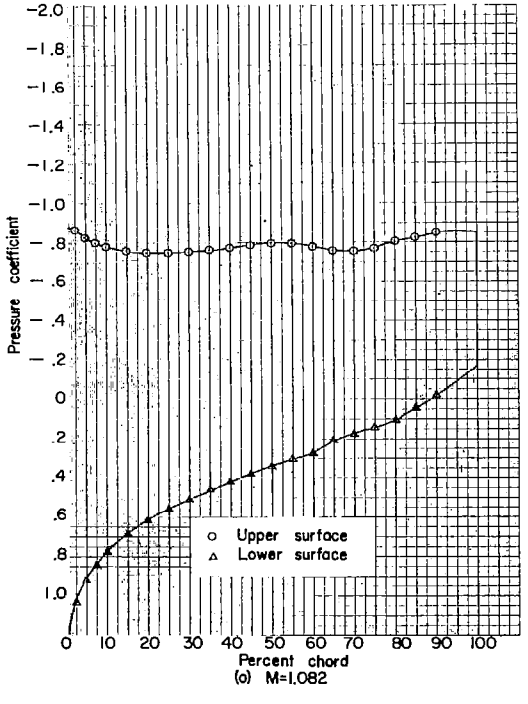


(l) M=1.006

Figure 18.- Continued. NACA 16-006; $\alpha = 12^{\circ}$.







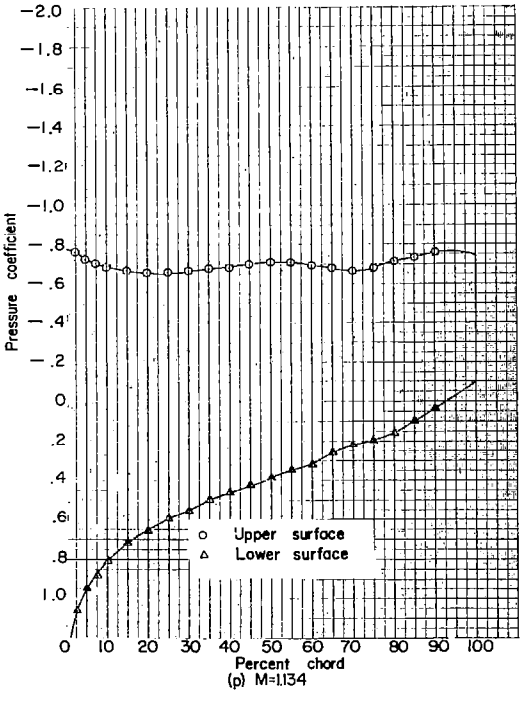


Figure 18.- Concluded. NACA 16-006; $\alpha = 12^{\circ}$.

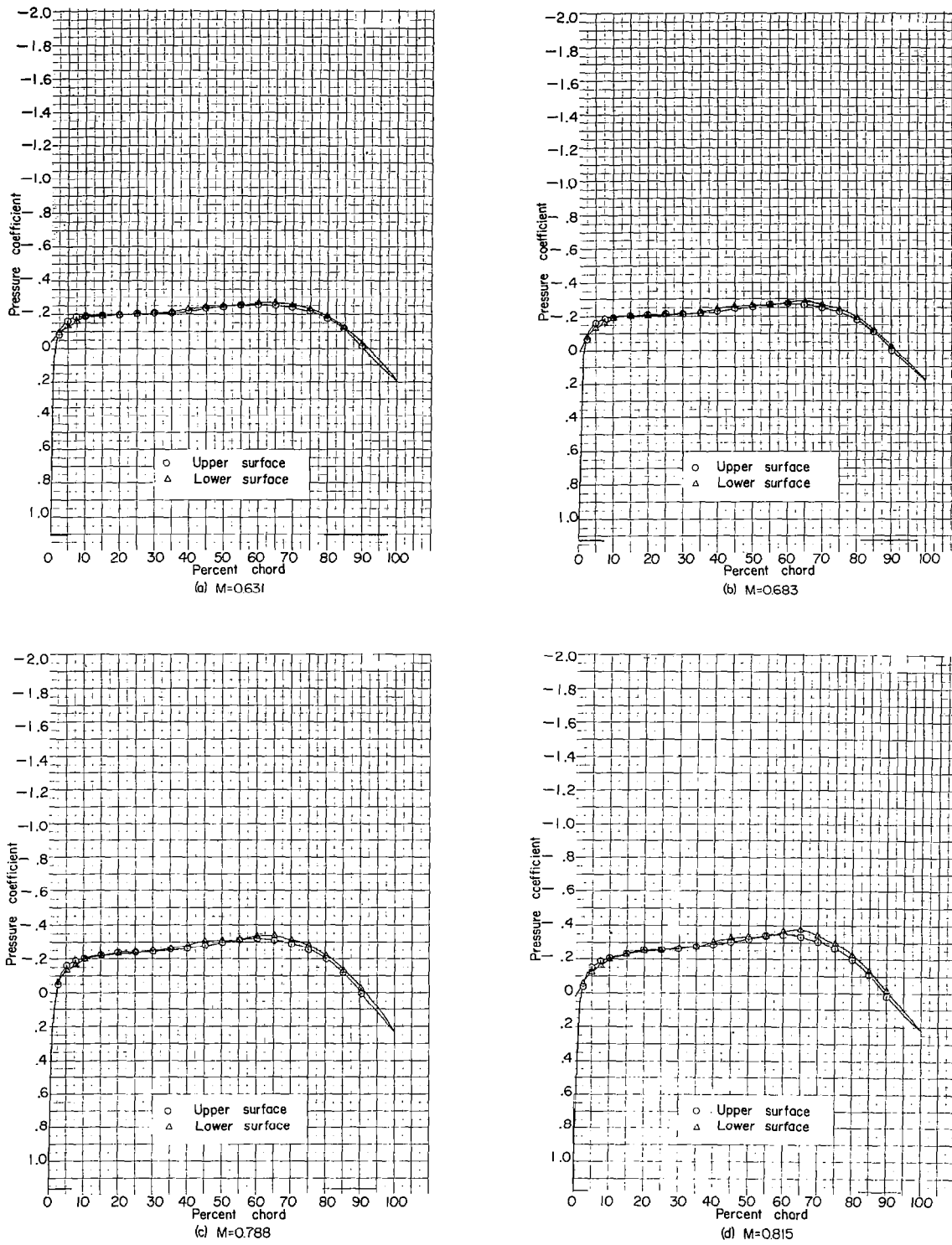
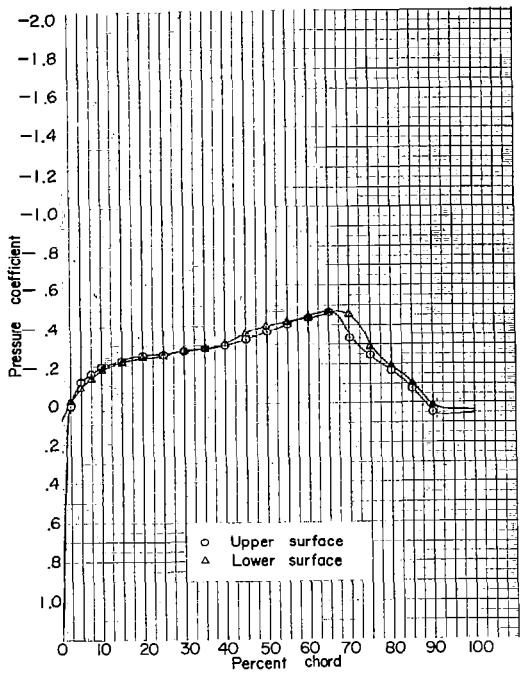
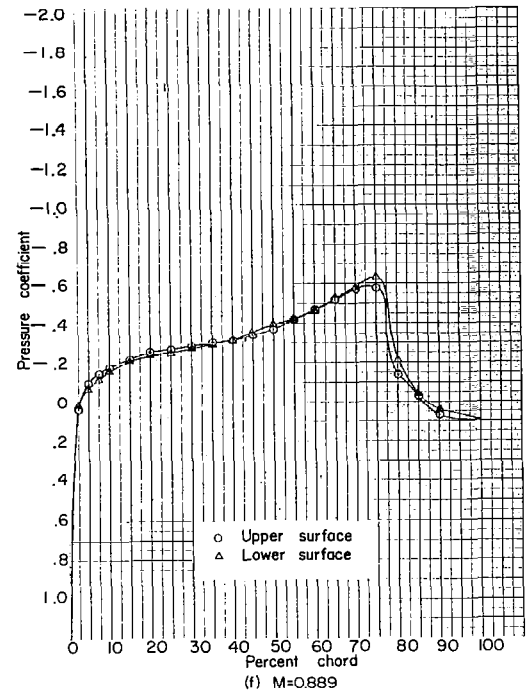


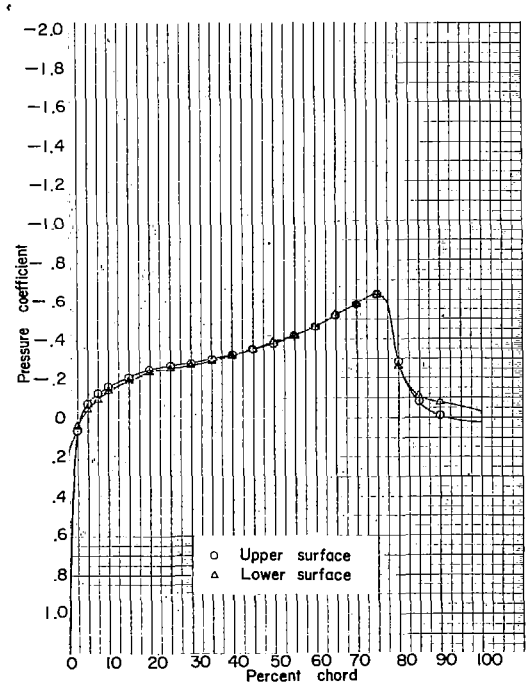
Figure 19.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 0^{\circ}$.



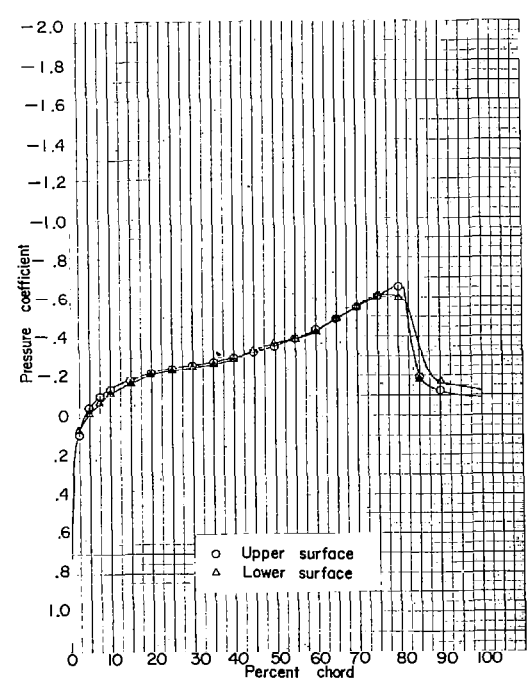
(e) M=0.863



(f) M=0.889

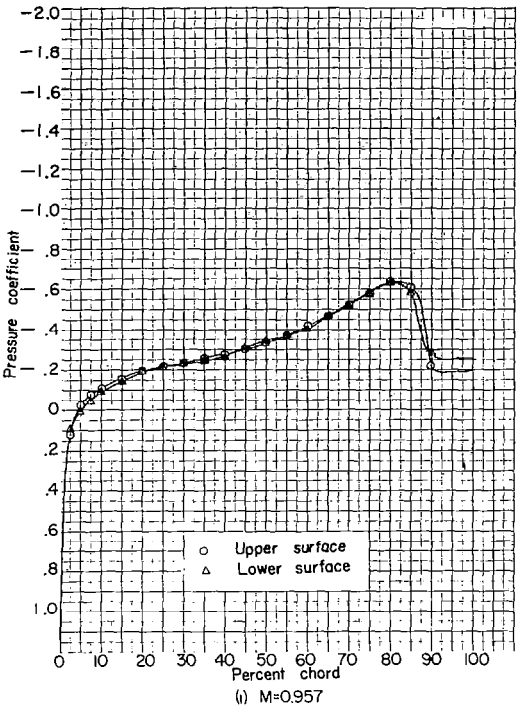


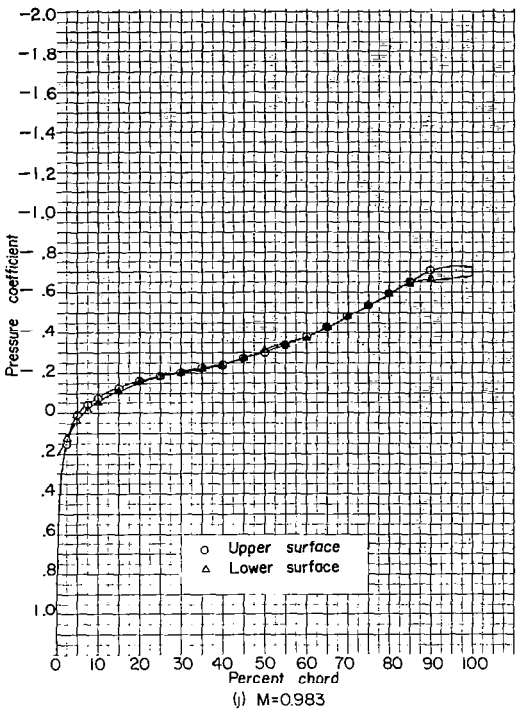
(g) M=0.914

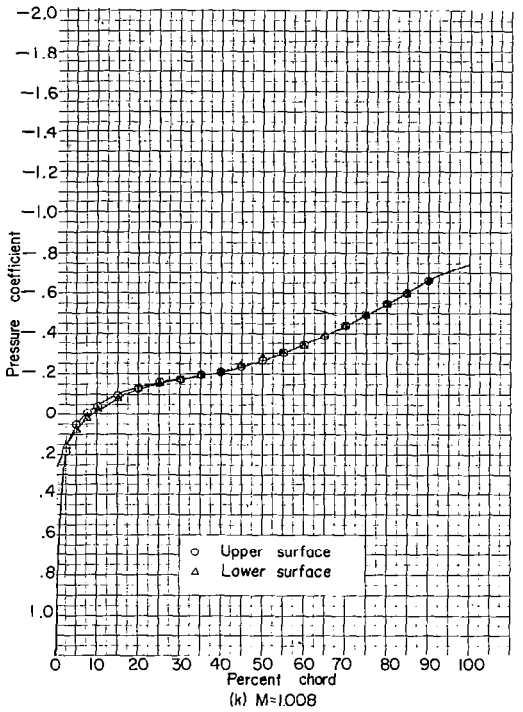


(h) M=0.939

Figure 19.- Continued. NACA 16-009; $\alpha = 0^{\circ}$.







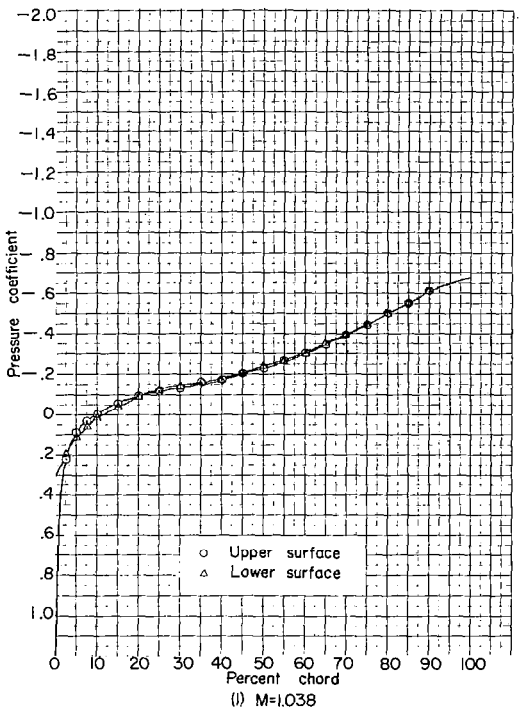
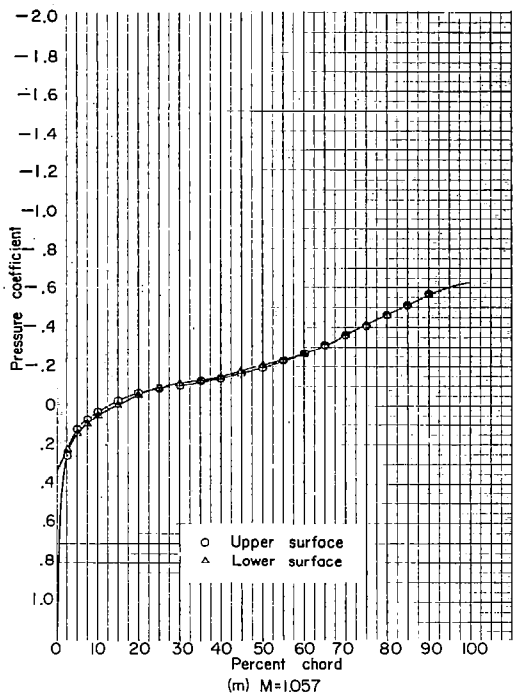
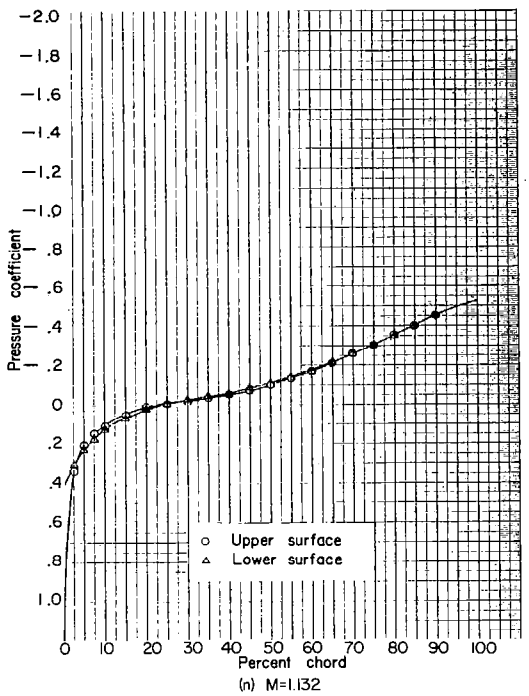
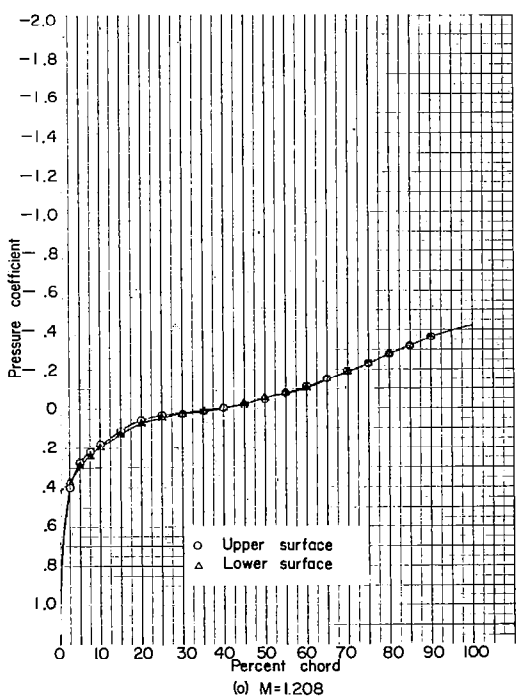


Figure 19.- Continued. NACA 16-009; $\alpha = 0^{\circ}$.







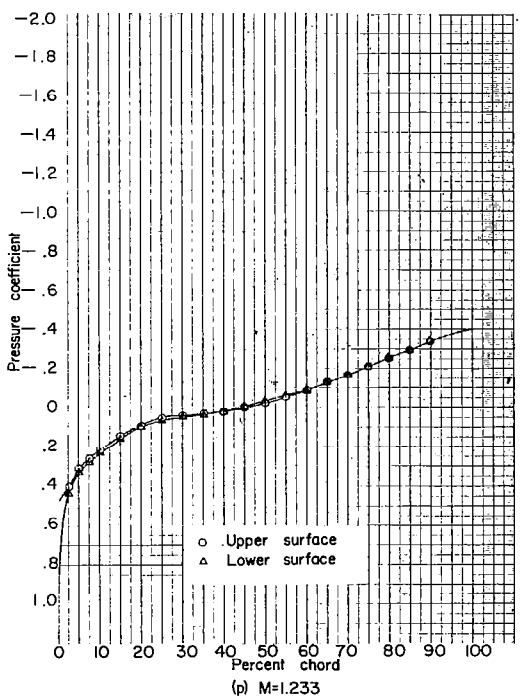


Figure 19.- Concluded. NACA 16-009; $\alpha = 0^{\circ}$.

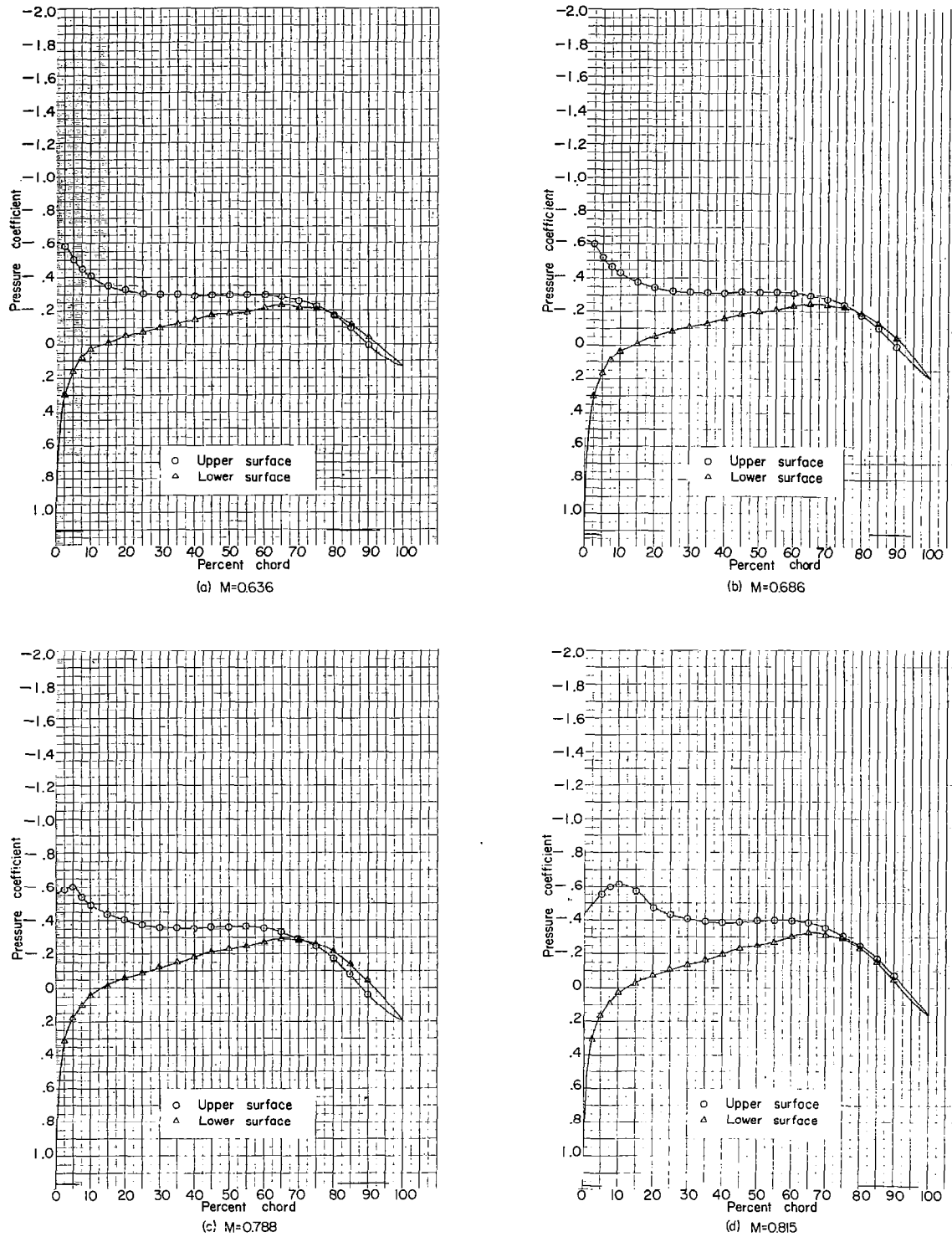
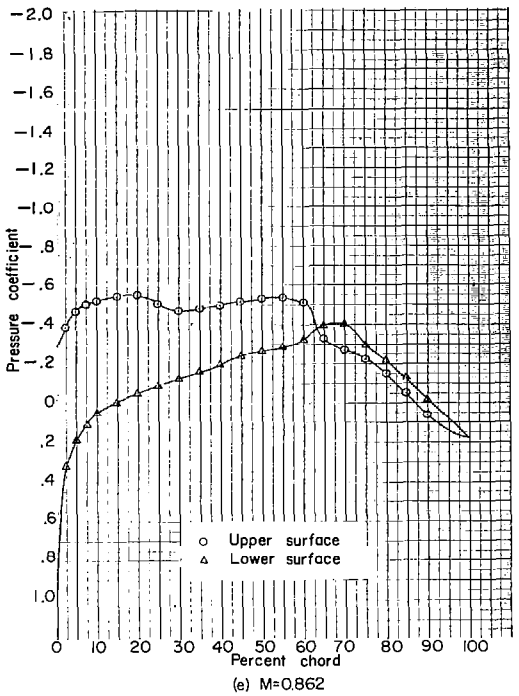
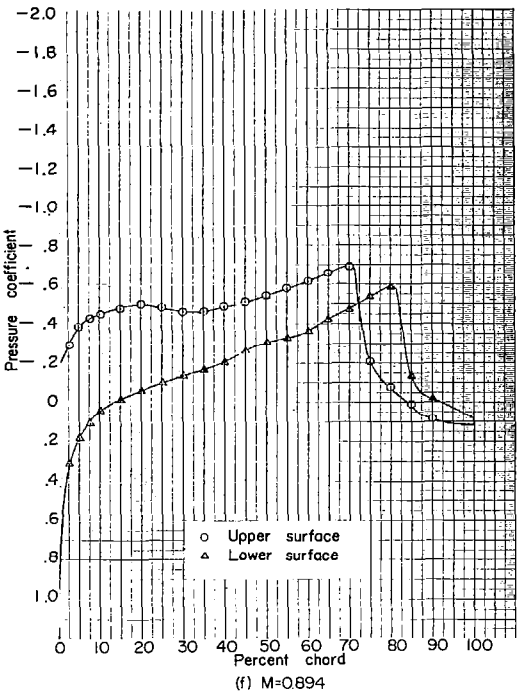


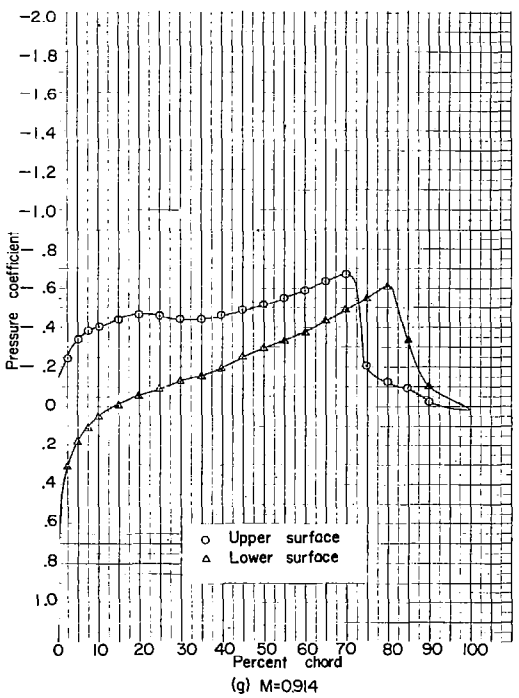
Figure 20.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 2^{\circ}$.



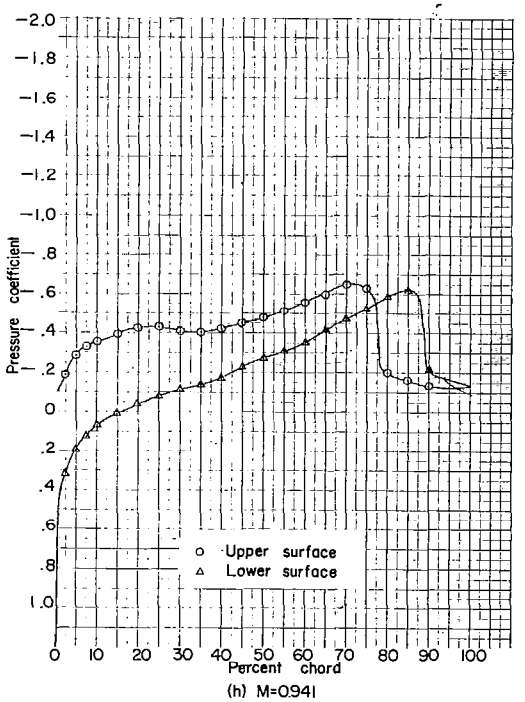
(e) M=0.862



(f) M=0.894

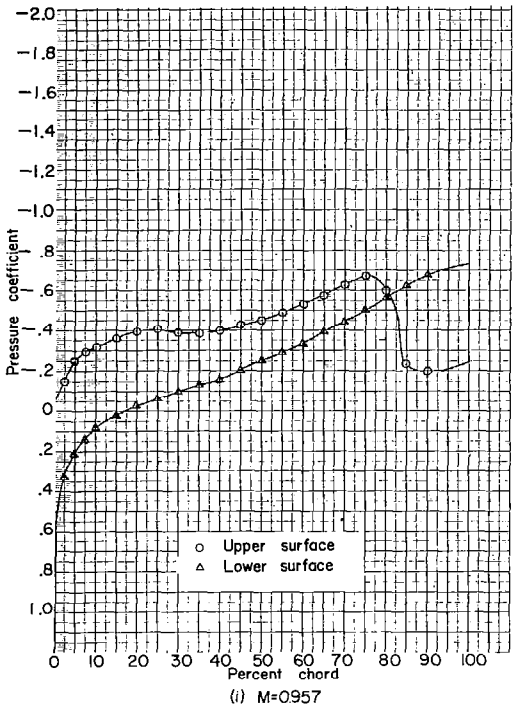


(g) M=0.914

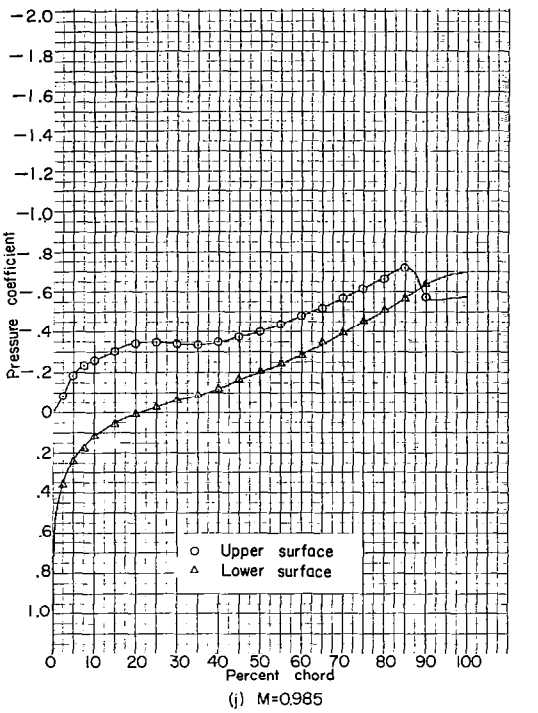


(h) M=0.941

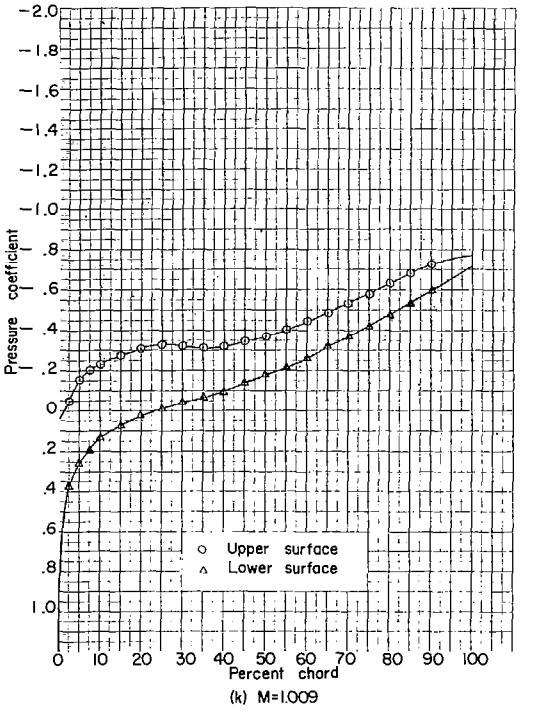
Figure 20.- Continued. NACA 16-009; $\alpha = 2^{\circ}$.



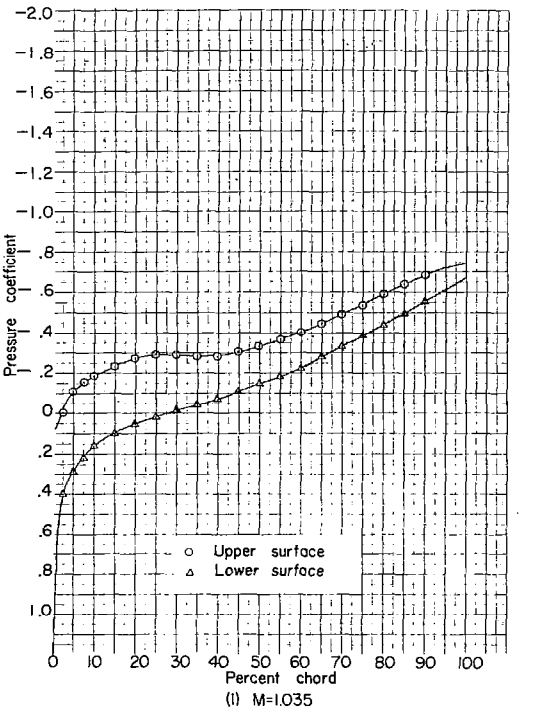
(i) M=0.957



(j) M=0.985

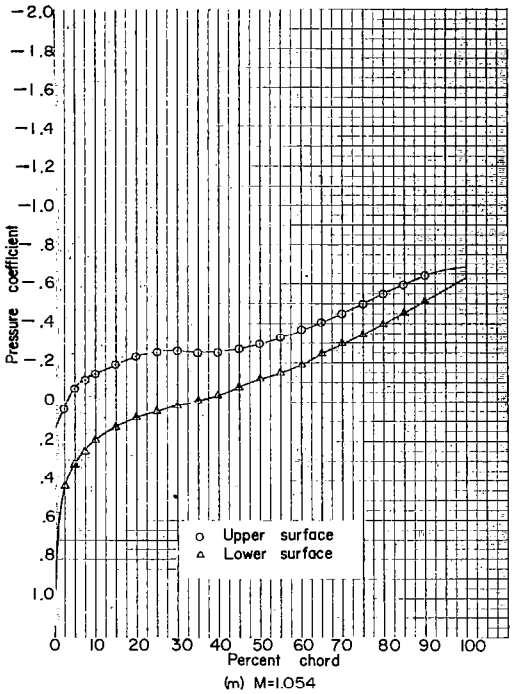


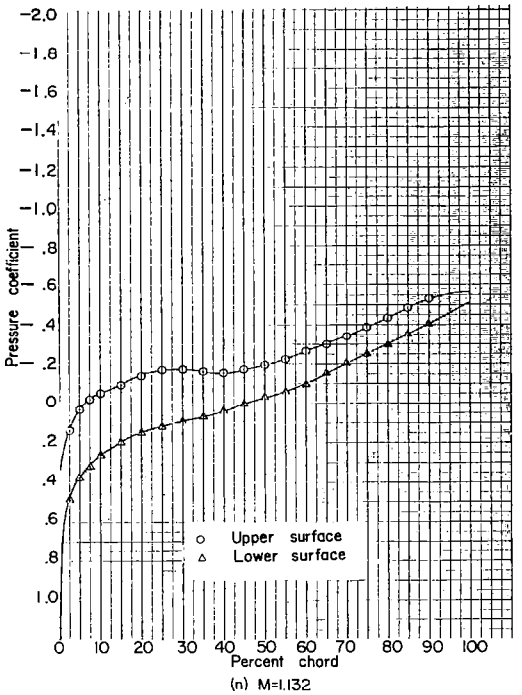
(k) M=1.009

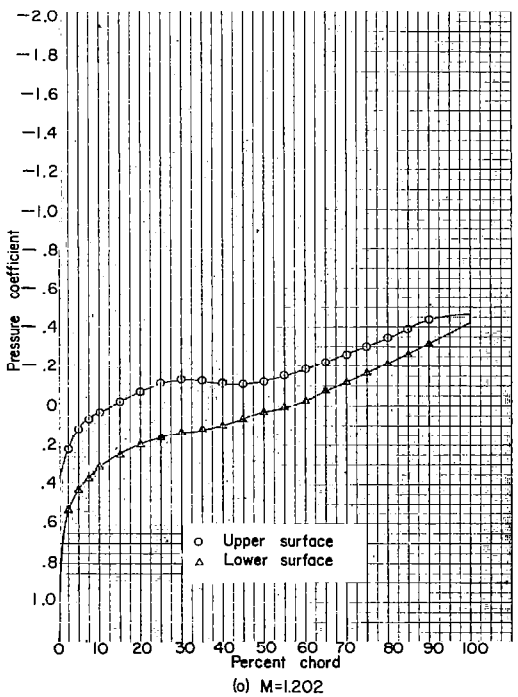


(l) M=1.035

Figure 20.- Continued. NACA 16-009; $\alpha = 2^{\circ}$.







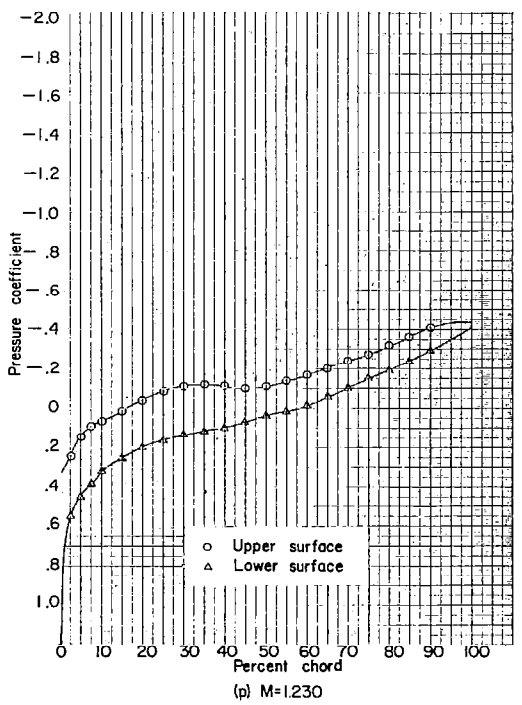


Figure 20.- Concluded. NACA 16-009; $\alpha = 2^{\circ}$.

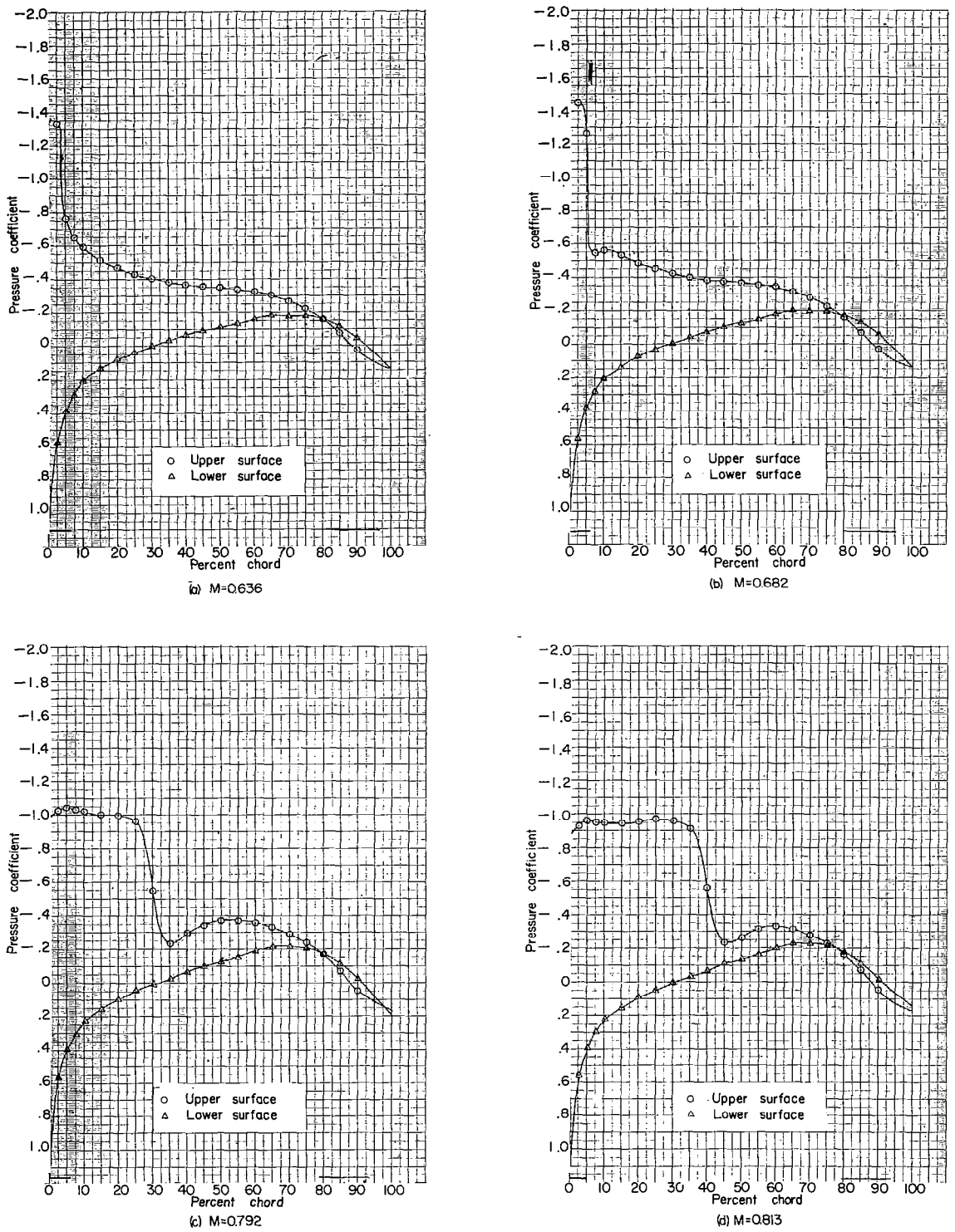
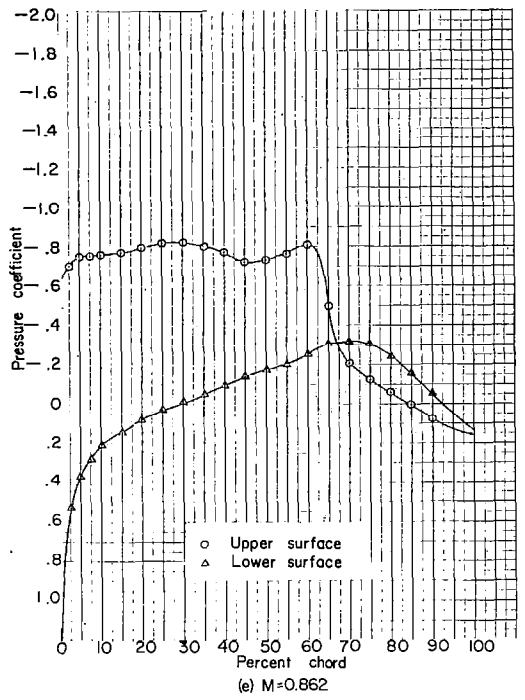
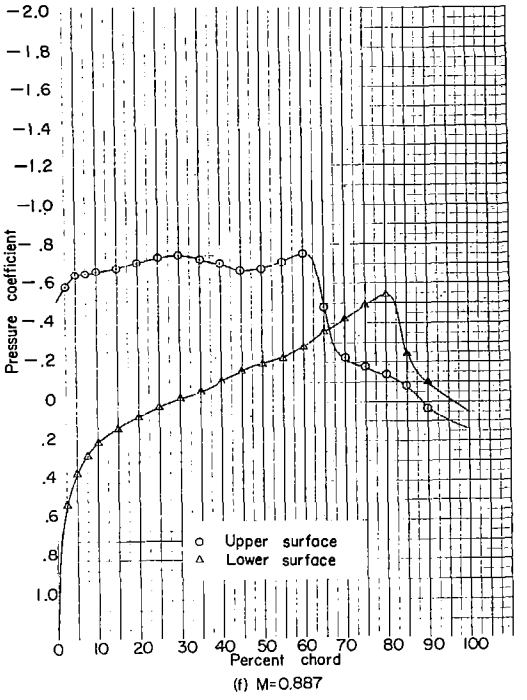


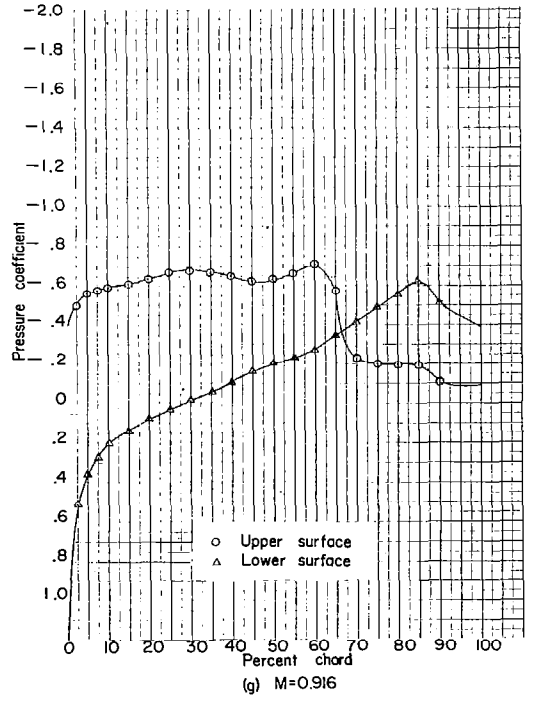
Figure 21.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 4^{\circ}$.



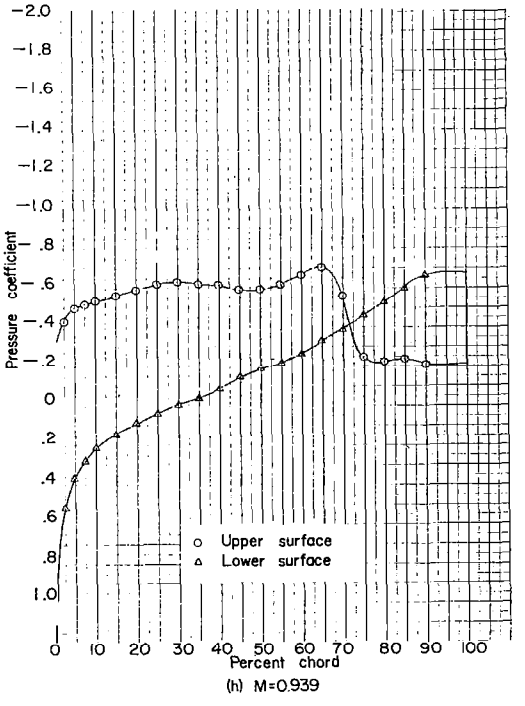
(e) M=0.862



(f) M=0.887

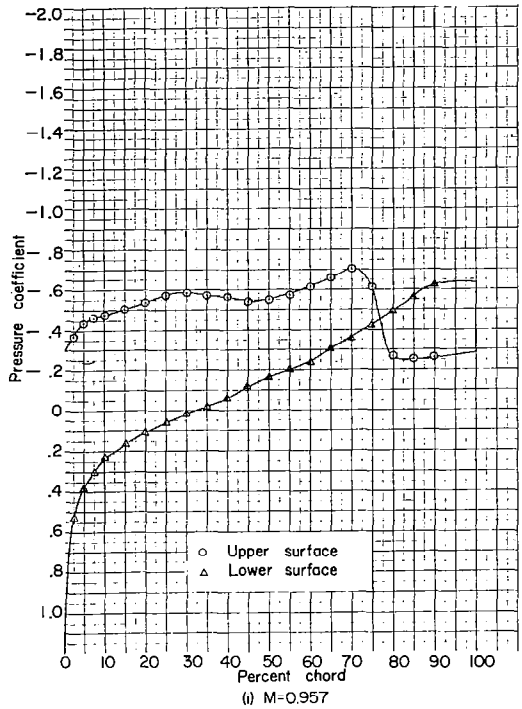


(g) M=0.916

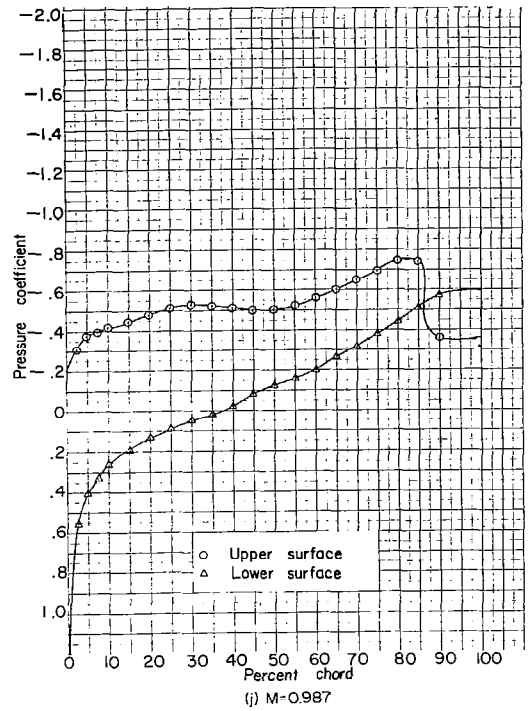


(h) M=0.939

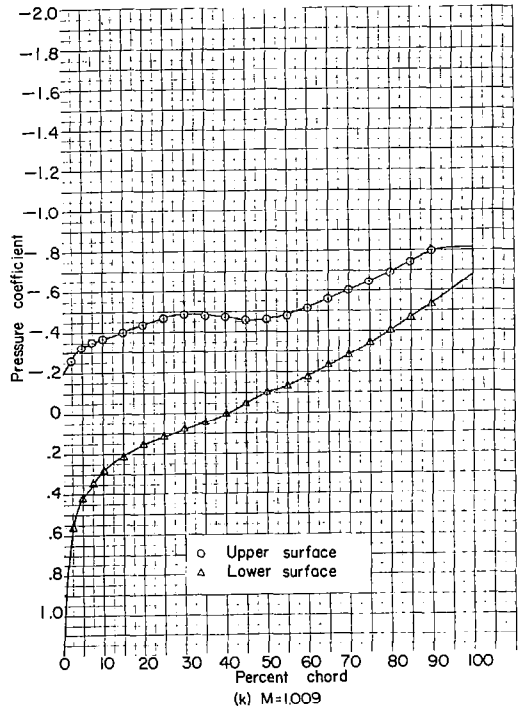
Figure 21.- Continued. NACA 16-009; $\alpha = 4^{\circ}$.



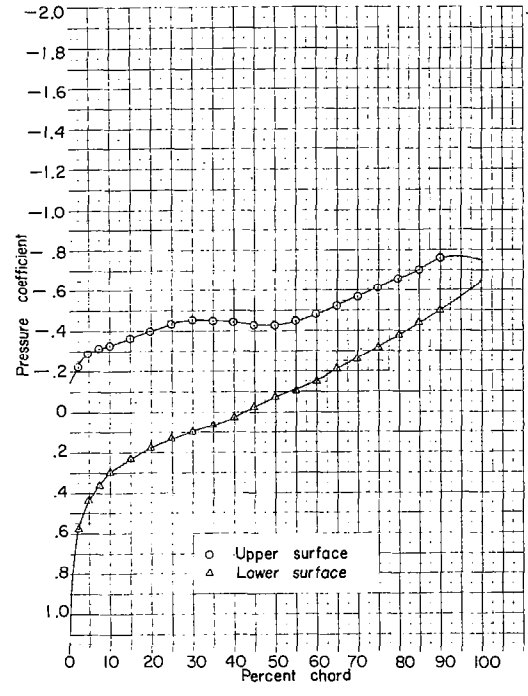
(i) M=0.957



(j) M=0.987

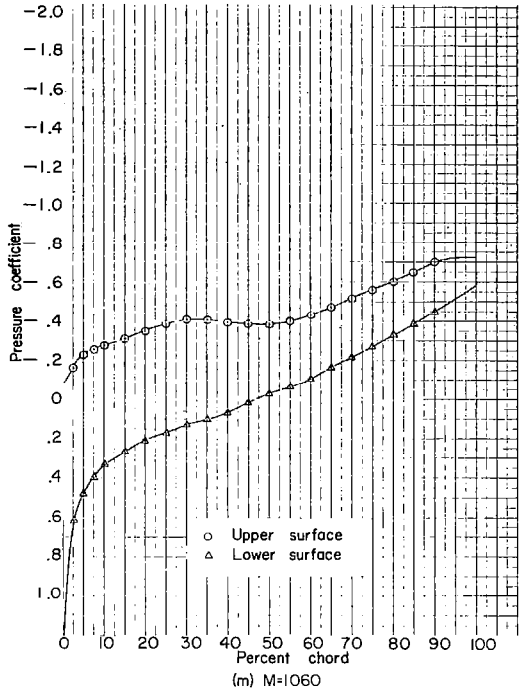


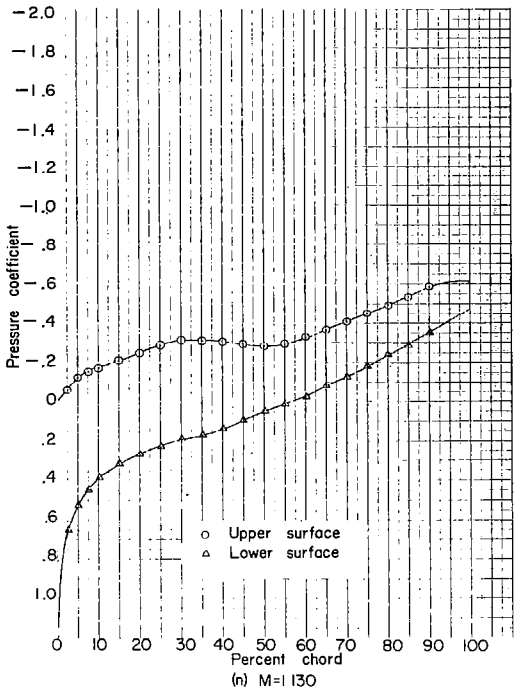
(k) M=1.009

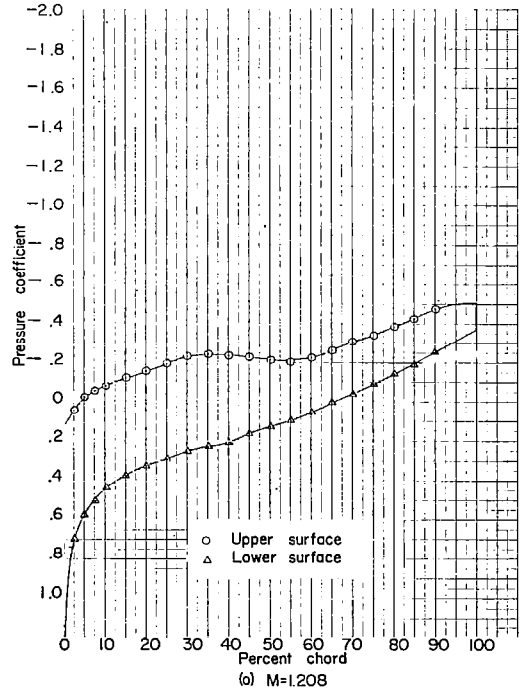


(l) M=1.033

Figure 21.- Continued. NACA 16-009; $\alpha = 4^{\circ}$.







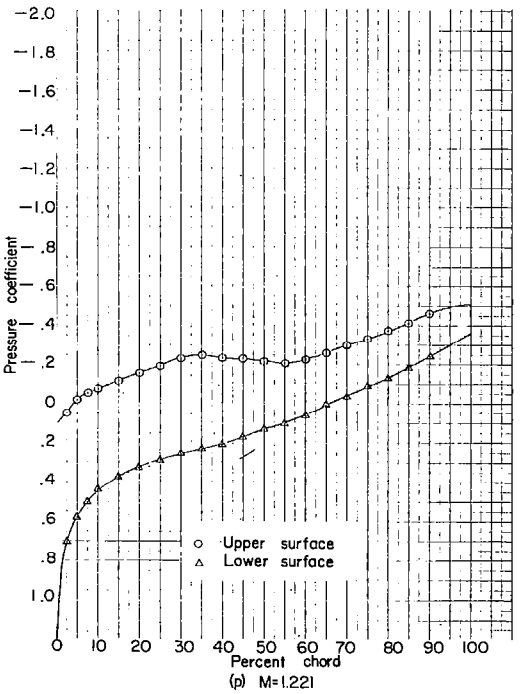


Figure 21.- Concluded. NACA 16-009; $\alpha = 4^{\circ}$.

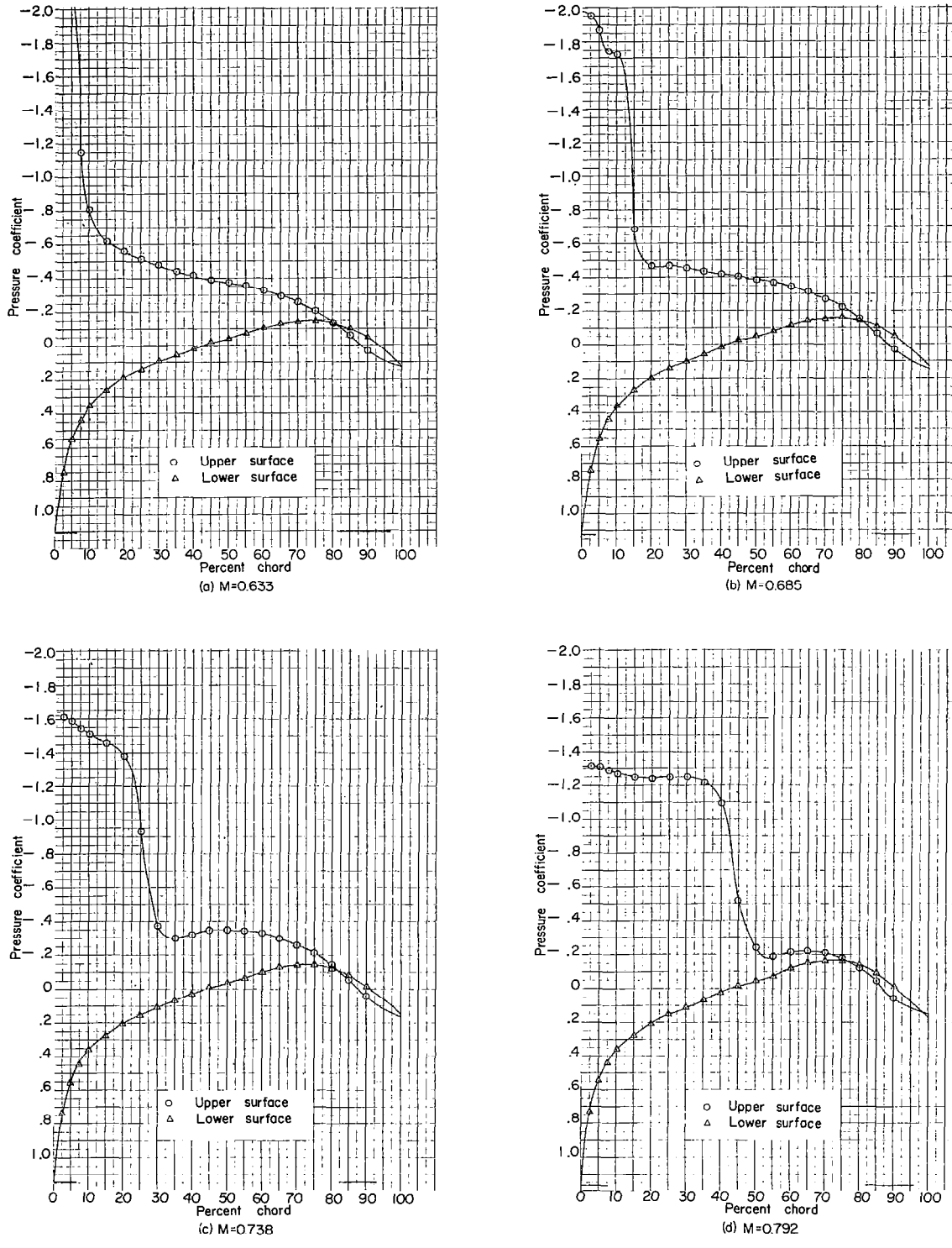
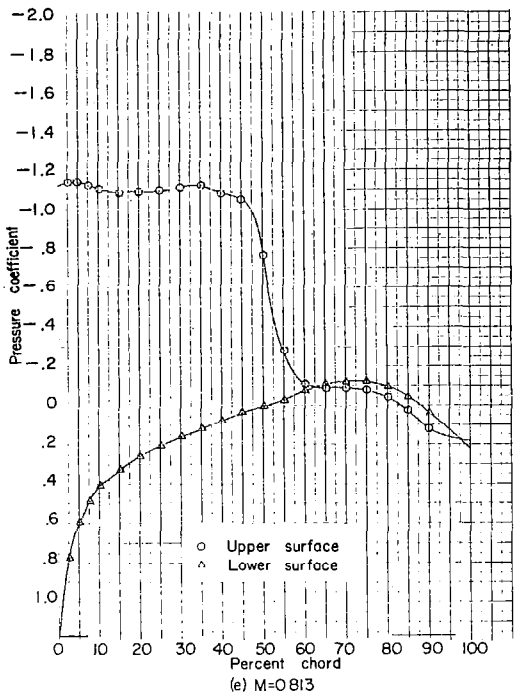
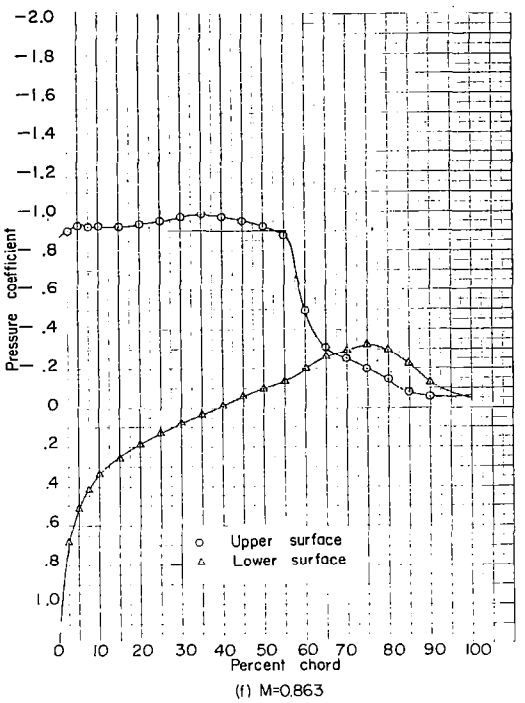


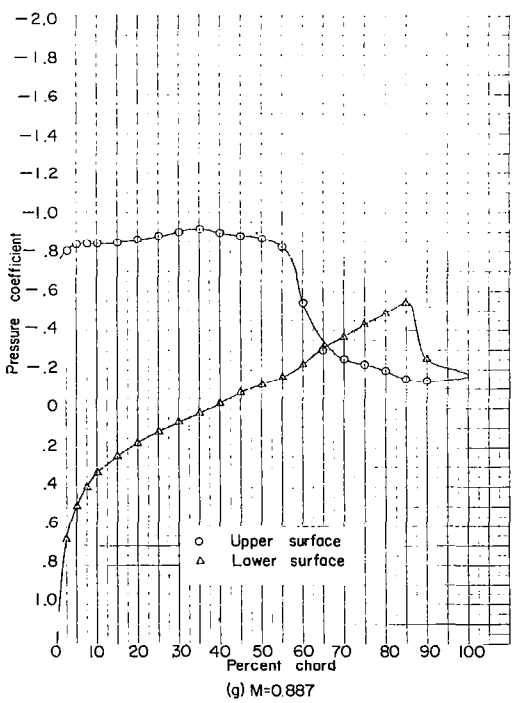
Figure 22.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 6^{\circ}$.



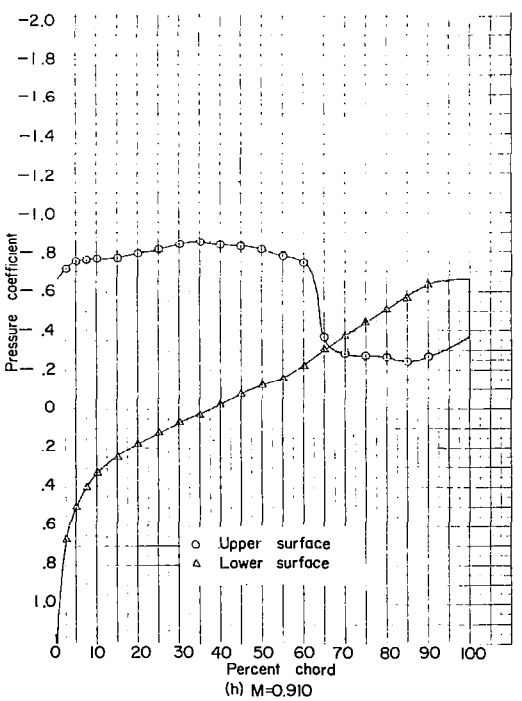
(e) M=0.813



(f) M=0.863

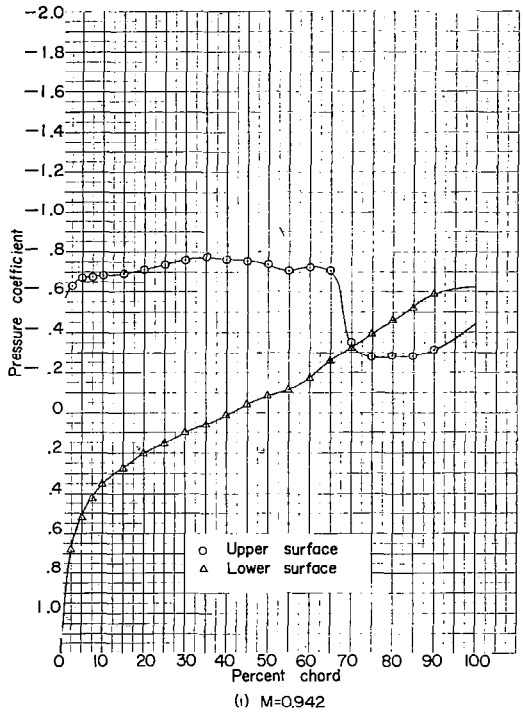


(g) M=0.887

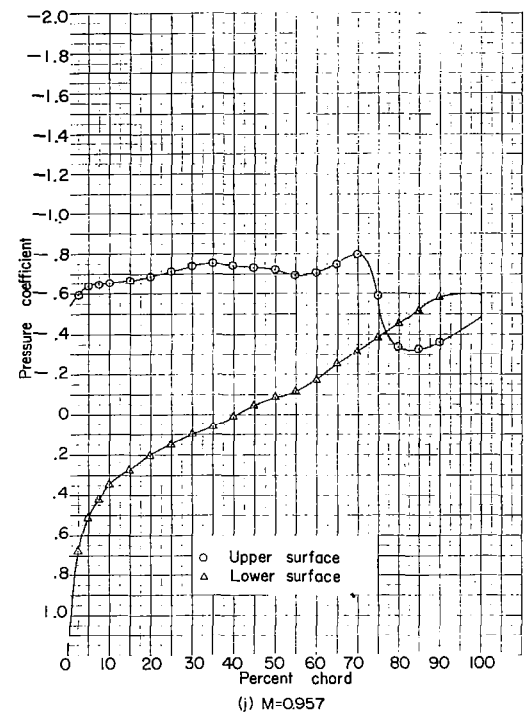


(h) M=0.910

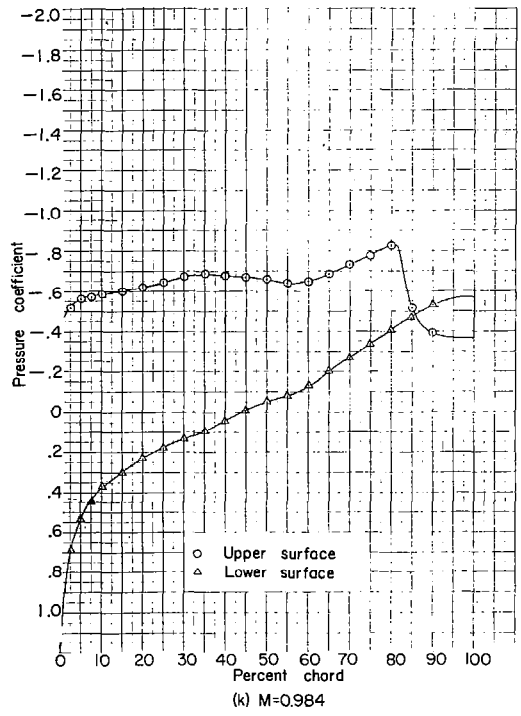
Figure 22.- Continued. NACA 16-009; $\alpha = 6^{\circ}$.



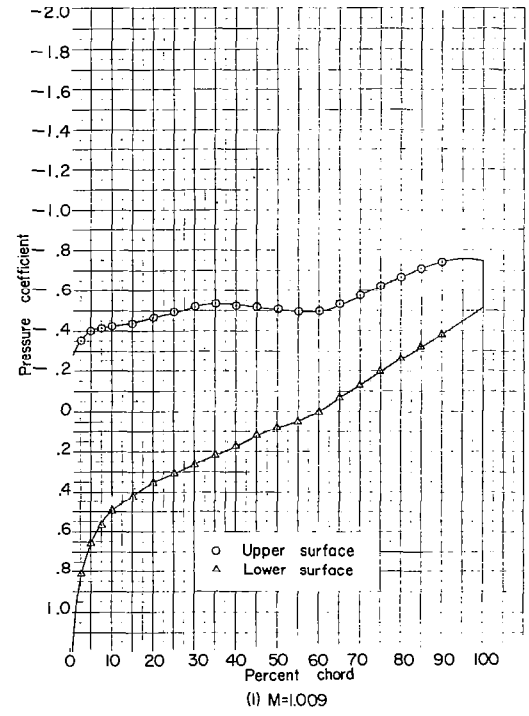
(i) M=0.942



(j) M=0.957

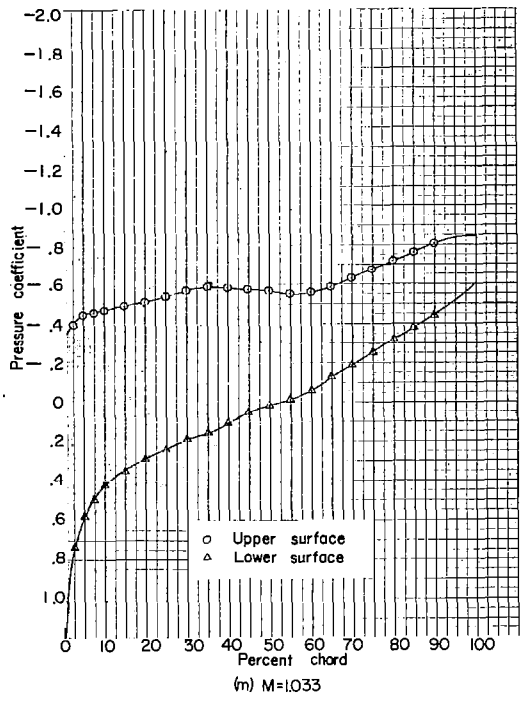


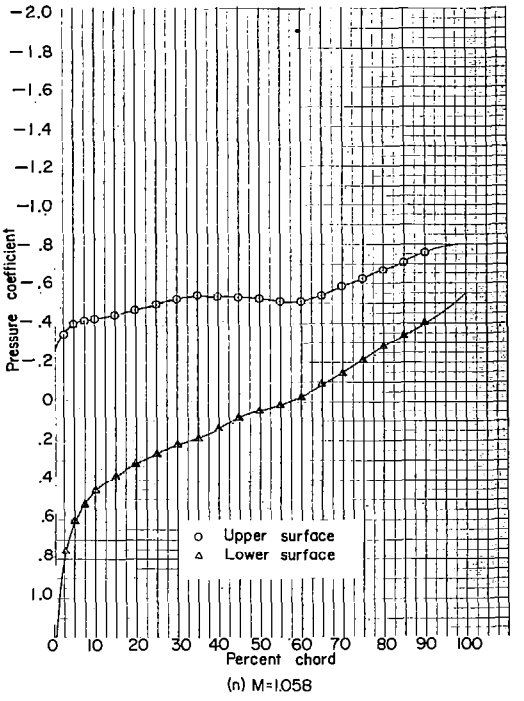
(k) M=0.984

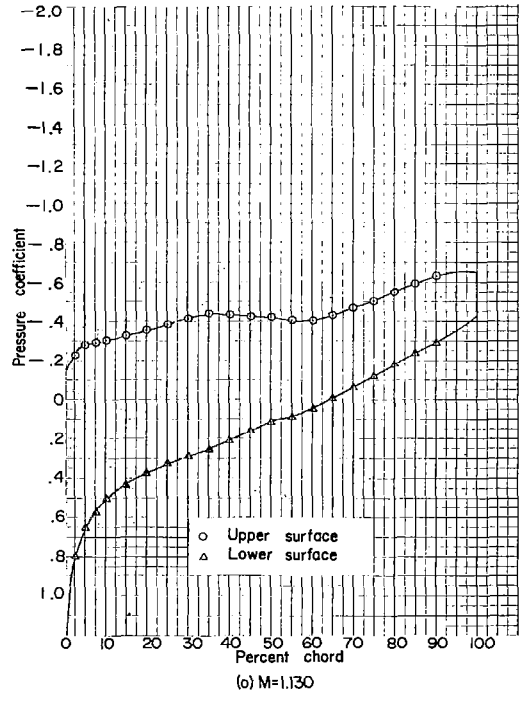


(l) M=1.009

Figure 22.- Continued. NACA 16-009; $\alpha = 6^{\circ}$.







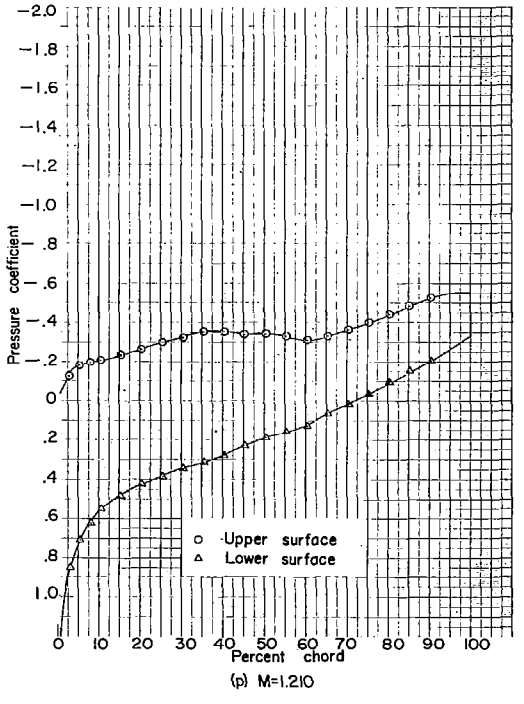


Figure 22.- Concluded. NACA 16-009; $\alpha = 6^{\circ}$.

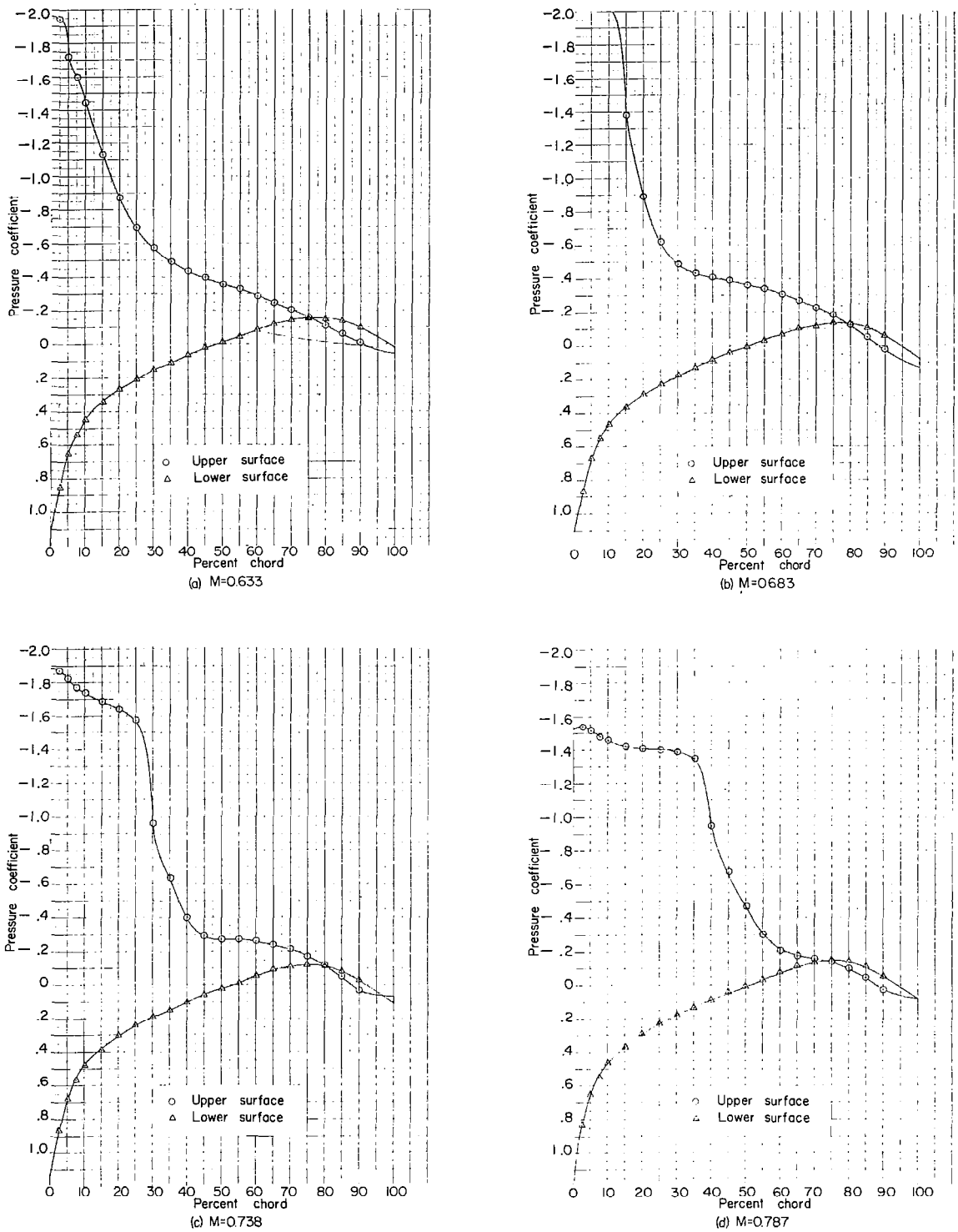
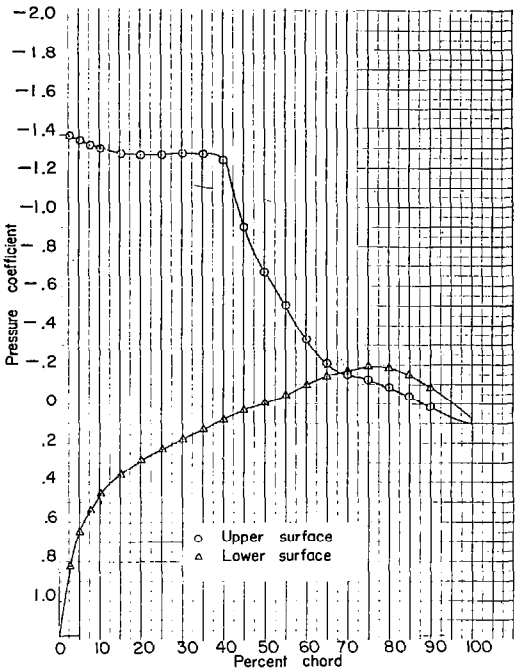
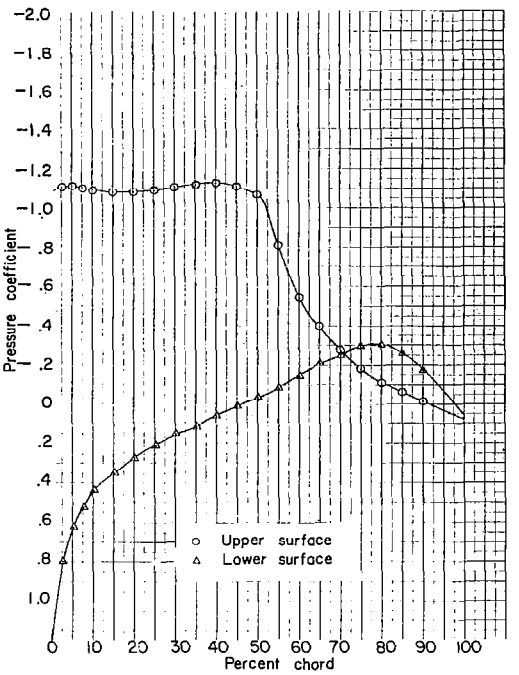


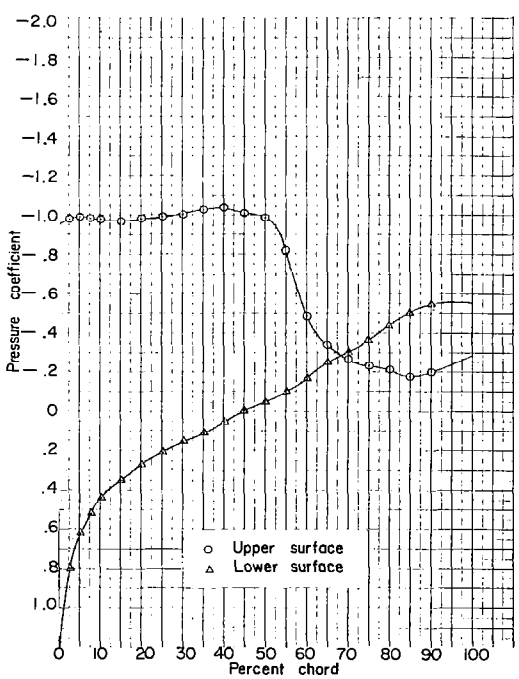
Figure 23.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 8^{o}$.



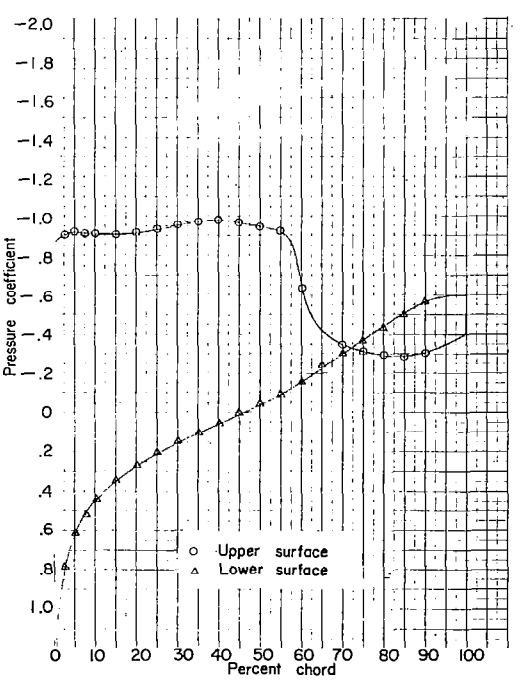
(e) M=0.813



(f) M=0.862

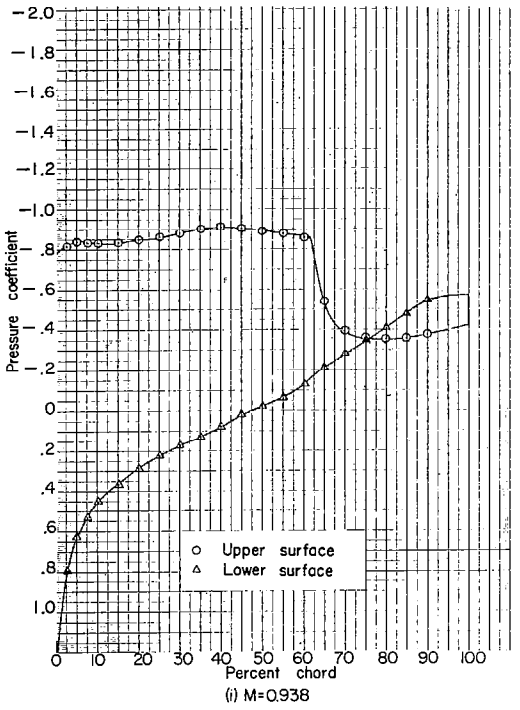


(g) M=0.889

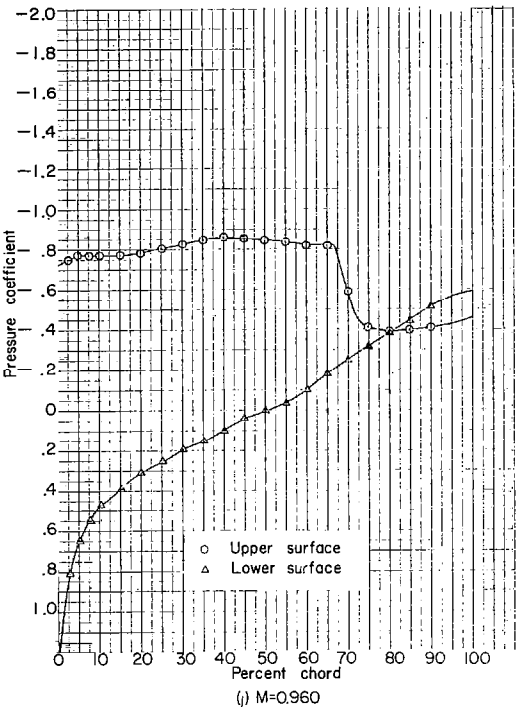


(h) M=0.913

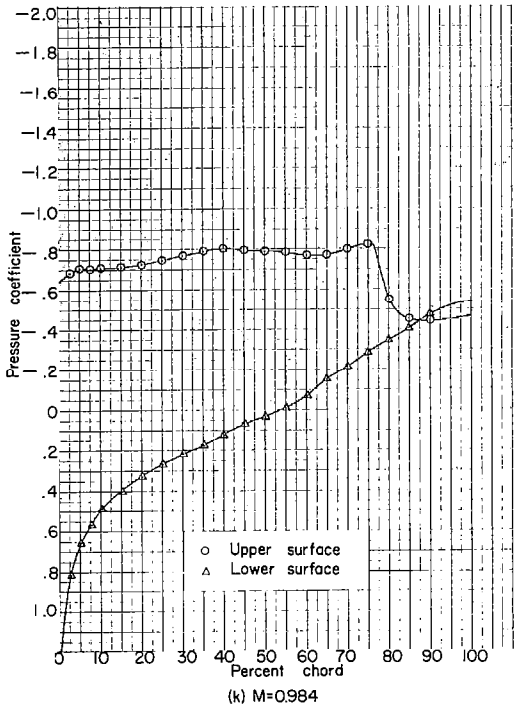
Figure 23.- Continued. NACA 16-009; $\alpha = 8^{\circ}$.



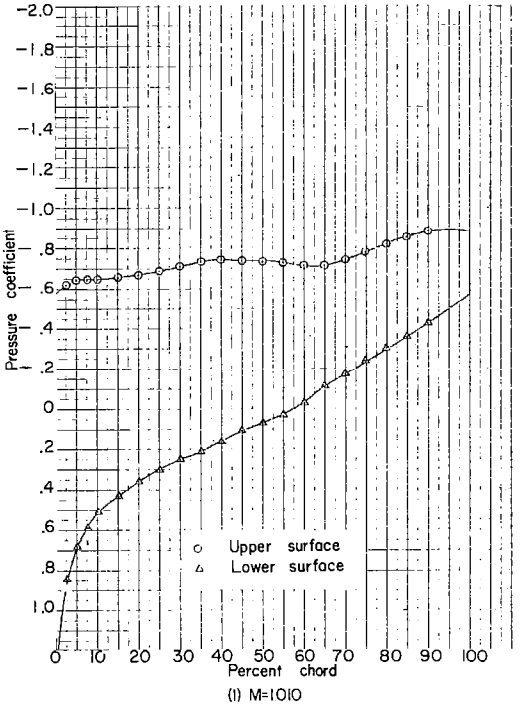
(i) M=0.938



(j) M=0.960

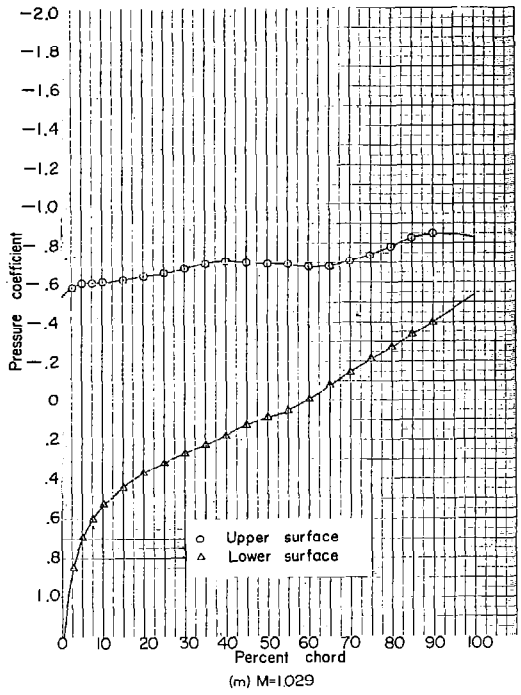


(k) M=0.984

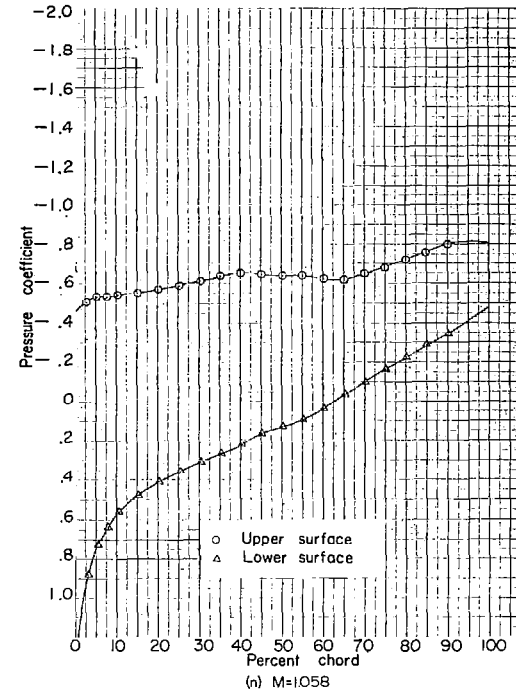


(l) M=1.010

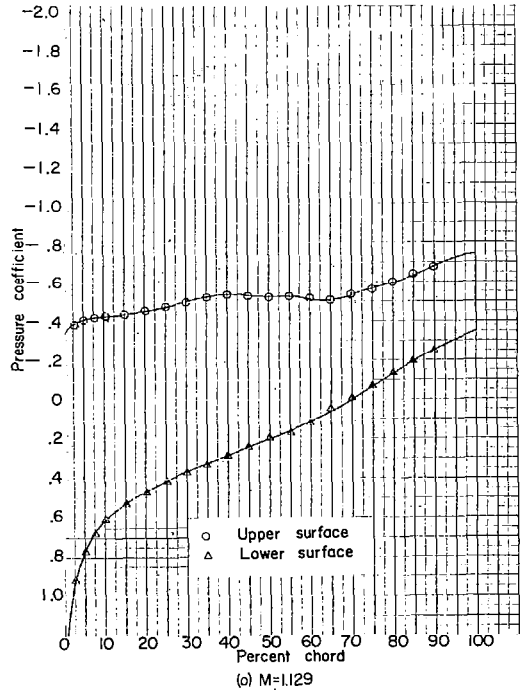
Figure 23.- Continued. NACA 16-009; $\alpha = 8^{\circ}$.



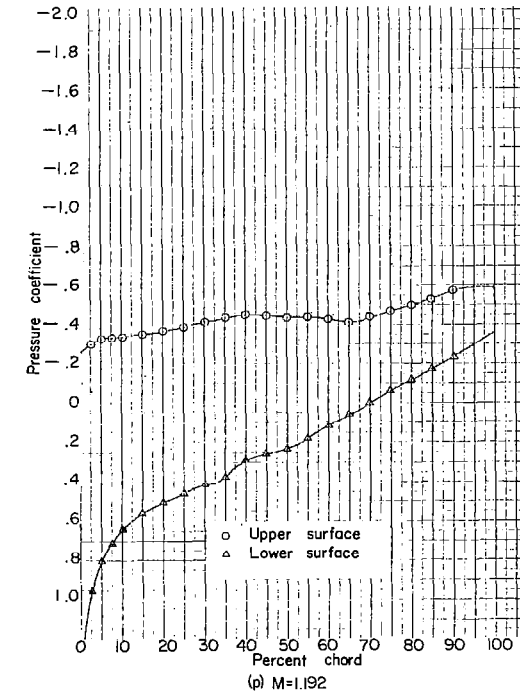
(m) M=1.029



(n) M=1.058



(o) M=1.129



(p) M=1.192

Figure 23.- Concluded. NACA 16-009; $\alpha = 8^{\circ}$.

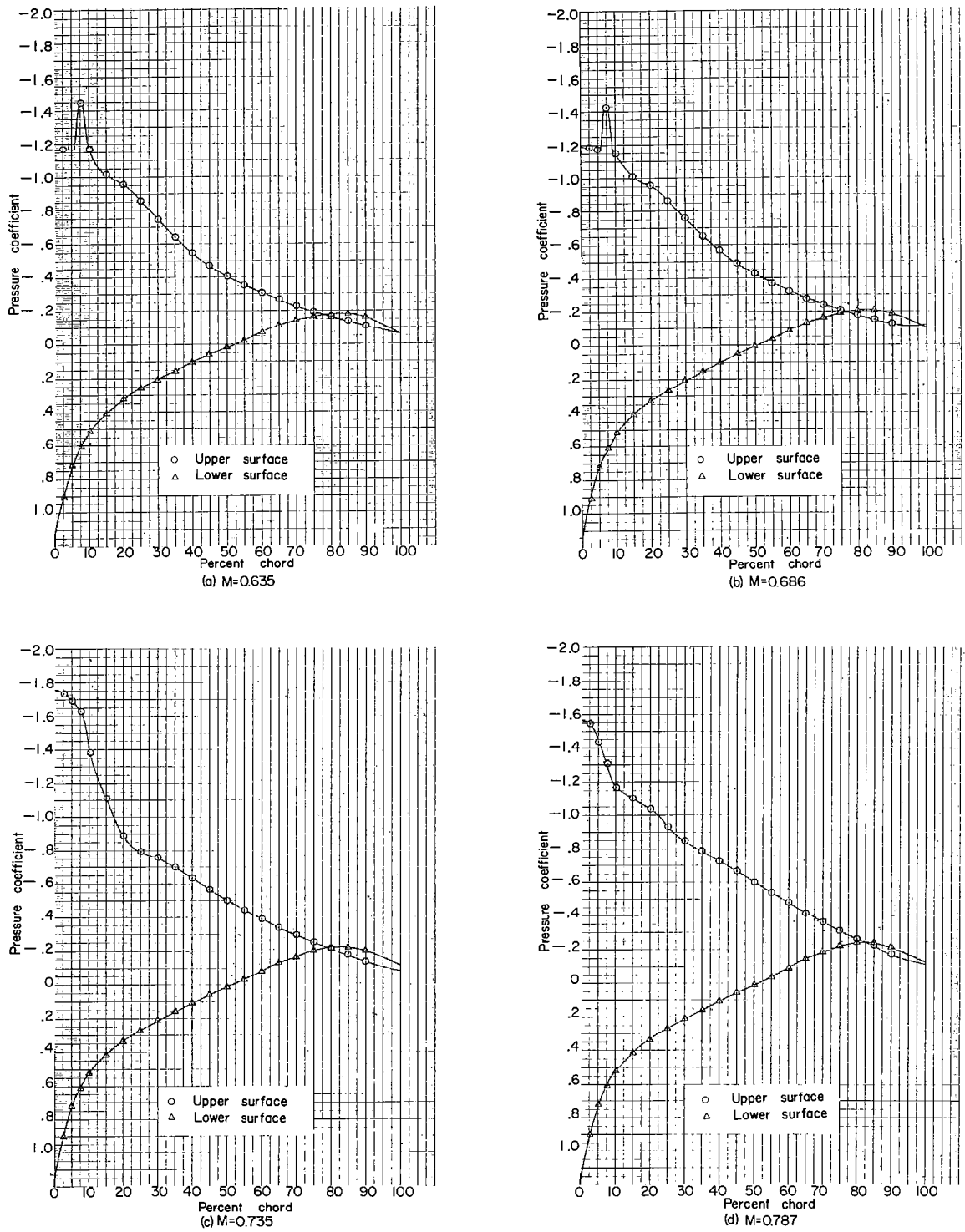
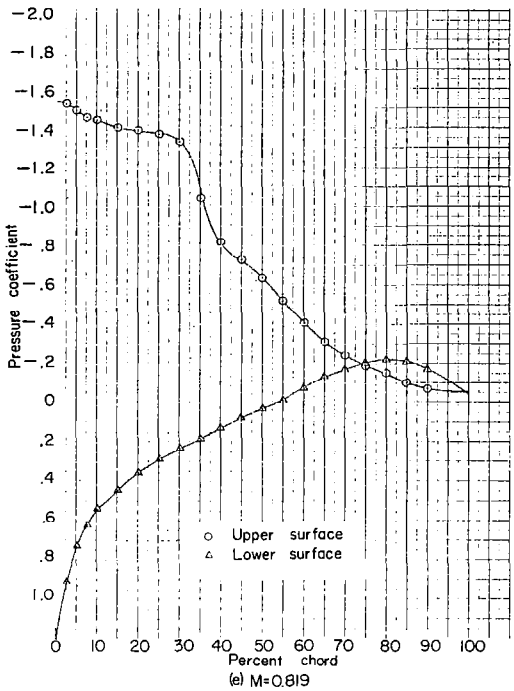
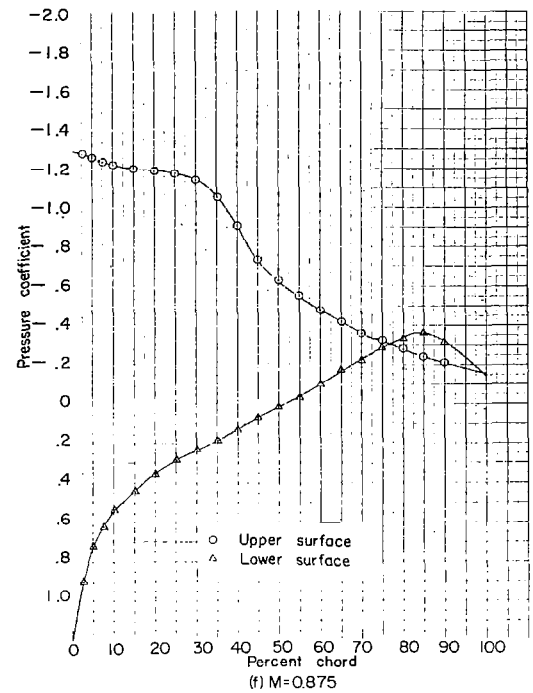
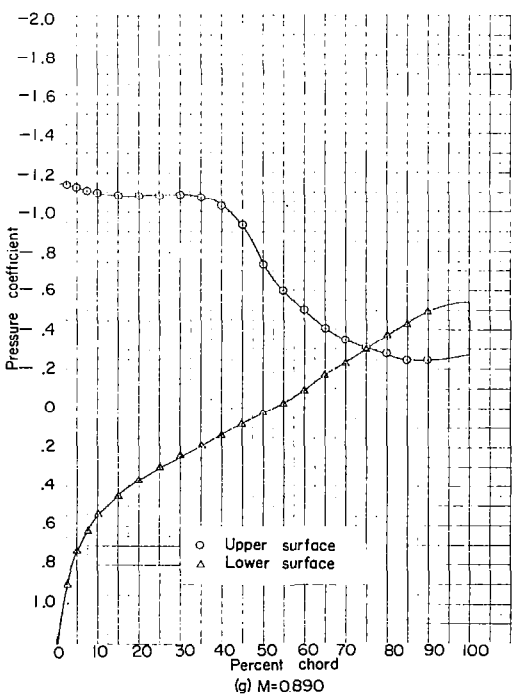


Figure 24.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 10^{\circ}$.







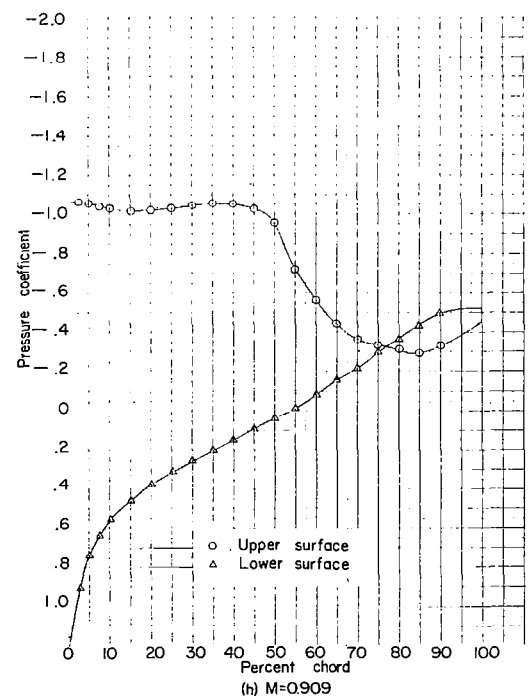
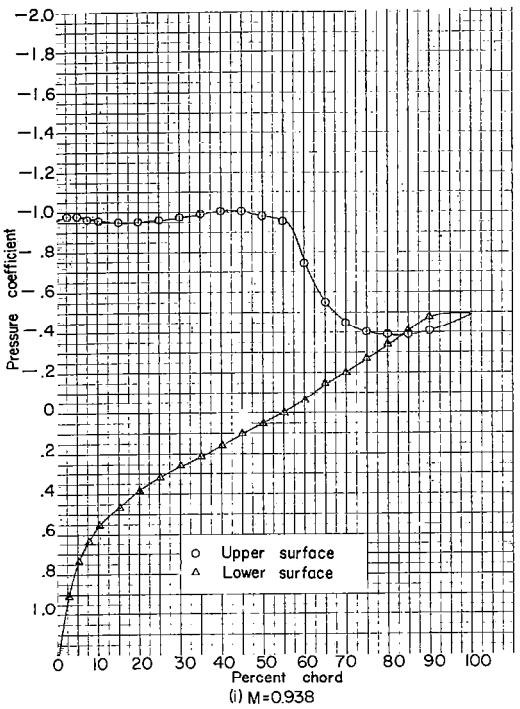
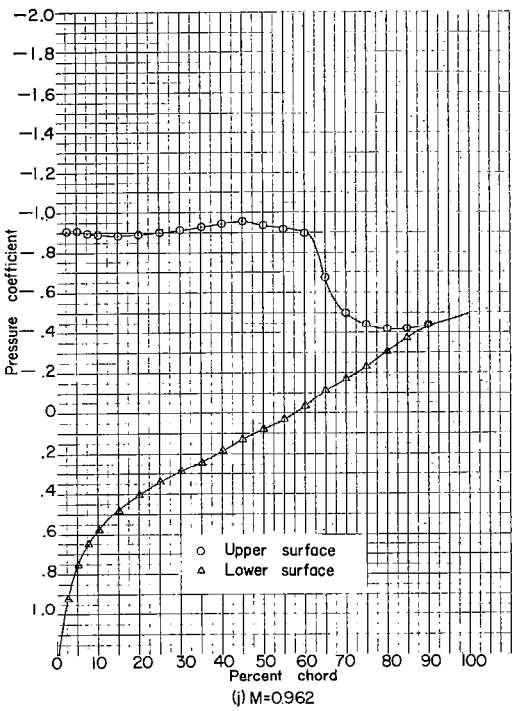
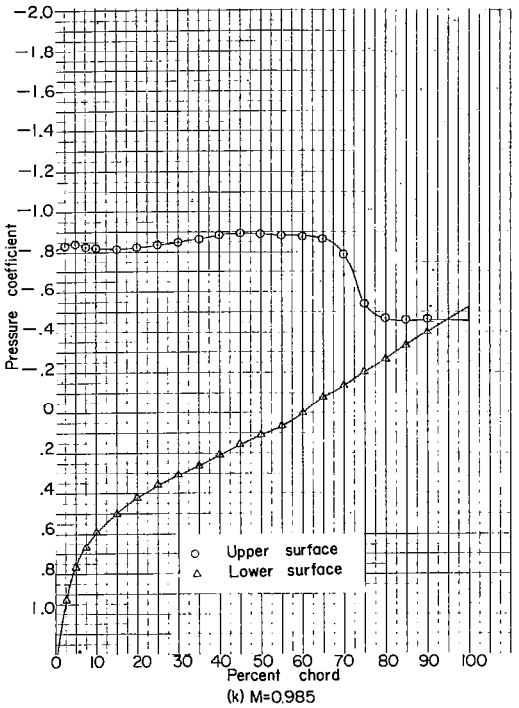


Figure 24.- Continued. NACA 16-009; $\alpha = 10^{\circ}$.







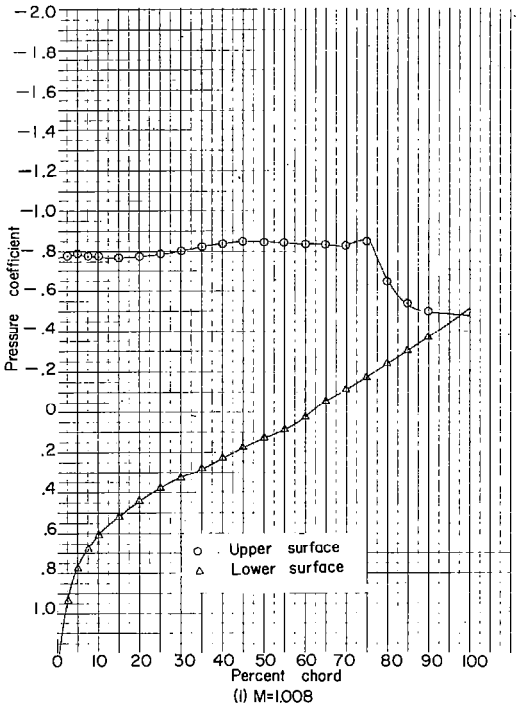
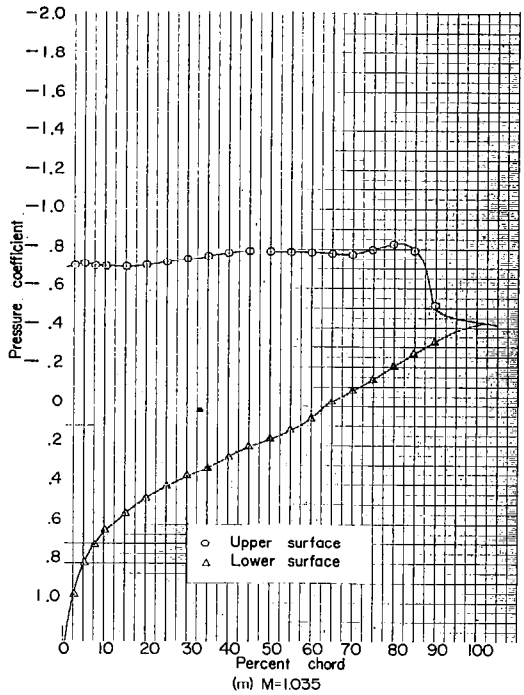
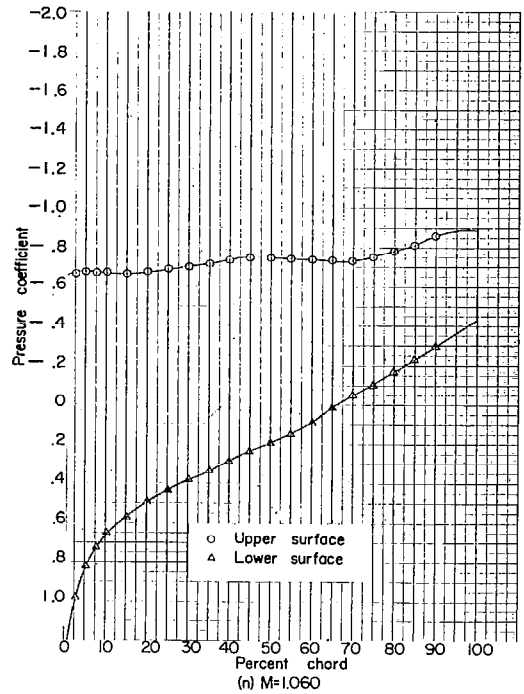
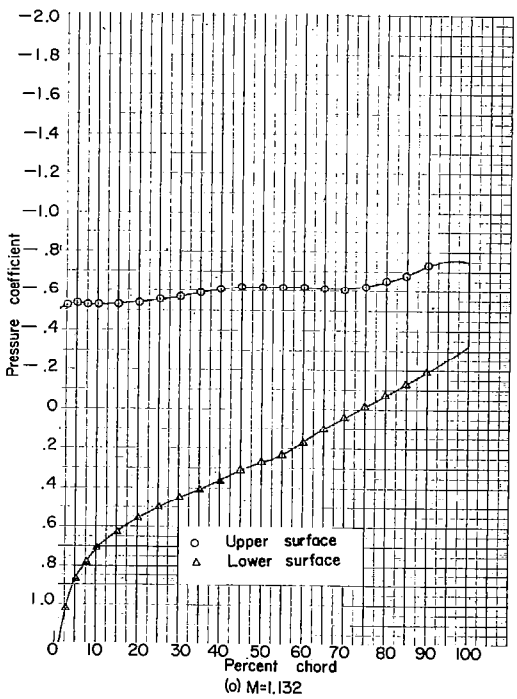


Figure 24.- Continued. NACA 16-009; $\alpha = 10^{\circ}$.







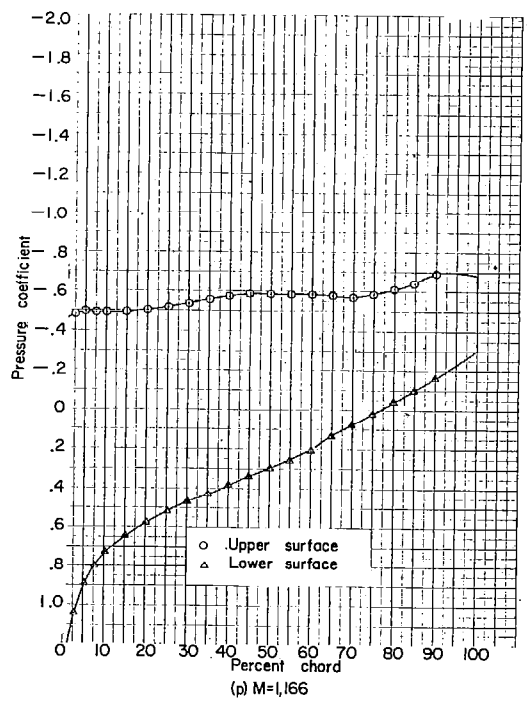


Figure 24.- Concluded. NACA 16-009; $\alpha = 10^{\circ}$.

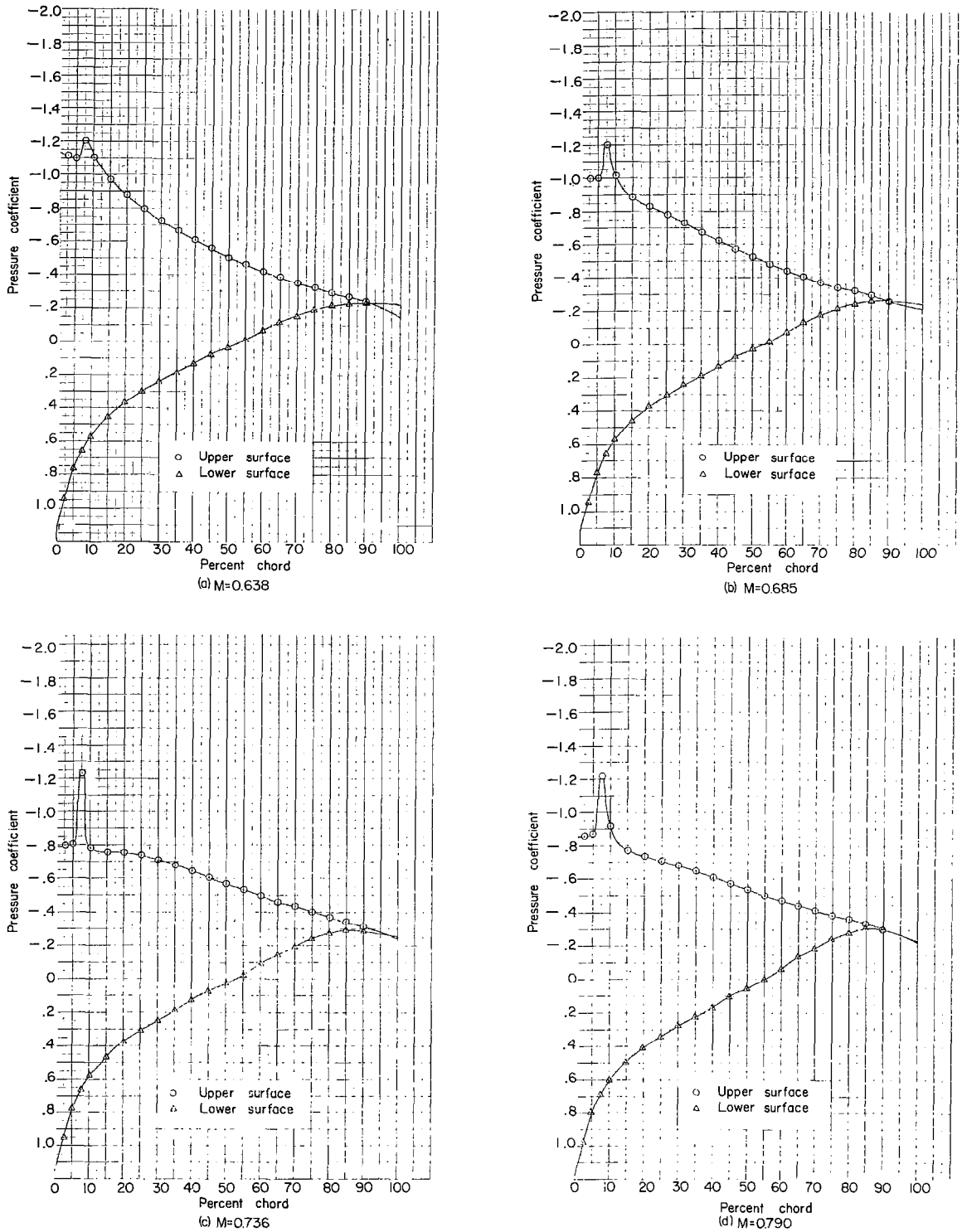
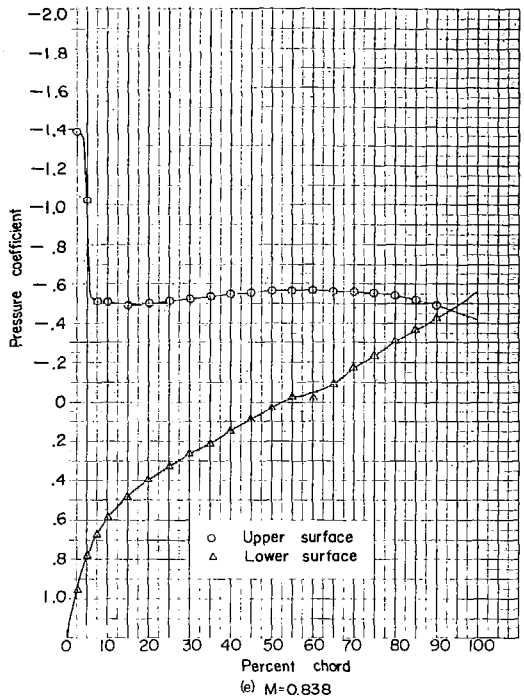
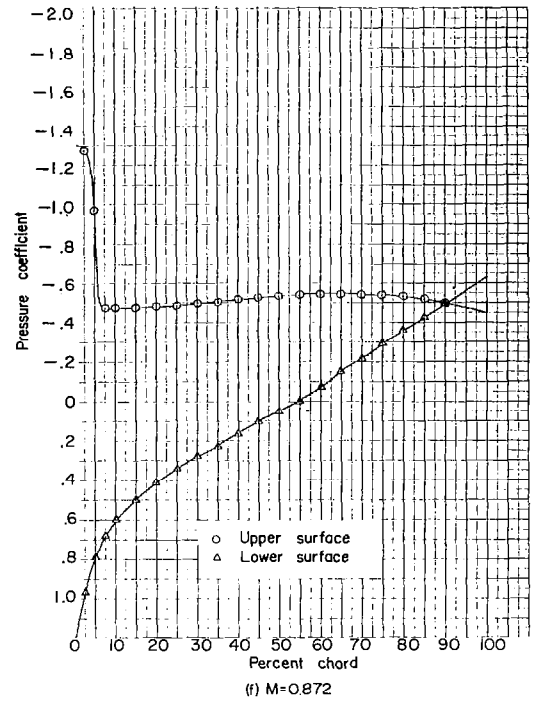


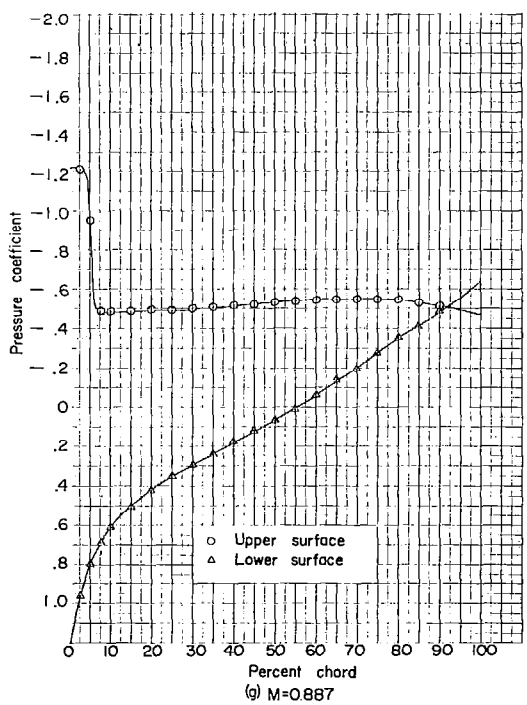
Figure 25.- Pressure distributions over NACA 16-009 airfoil section. $\alpha = 12^{\circ}$.



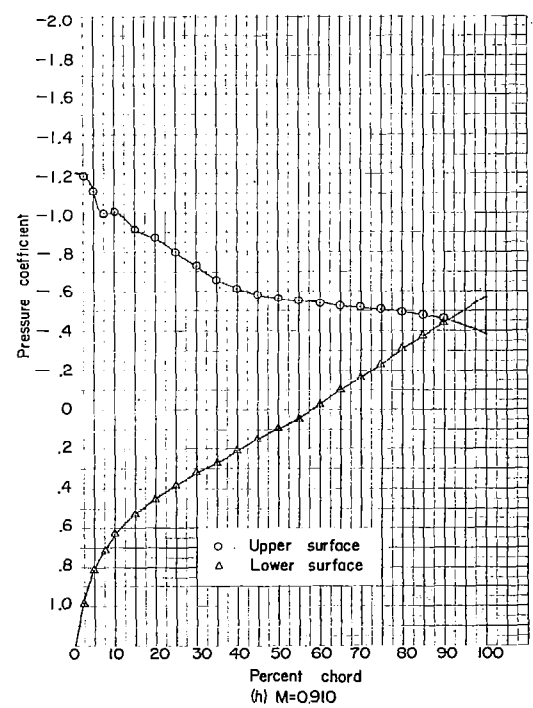
(e) M=0.838



(f) M=0.872

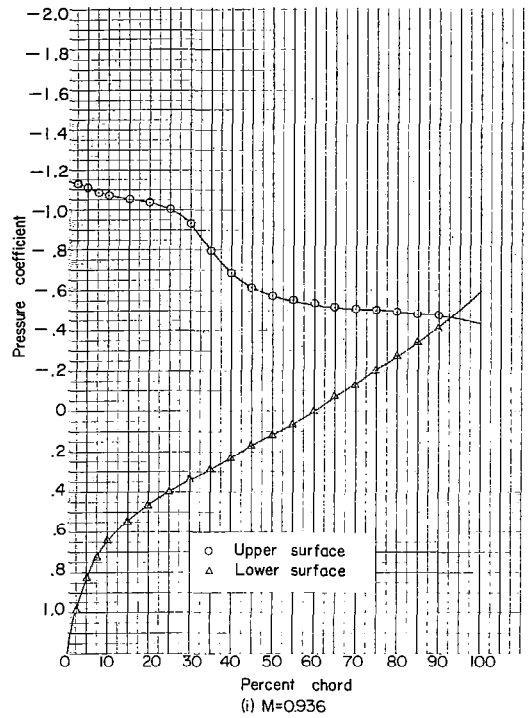


(g) M=0.887

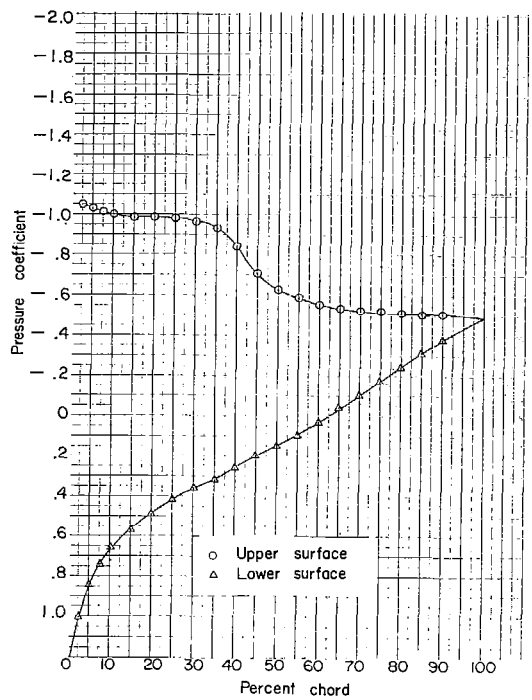


(h) M=0.910

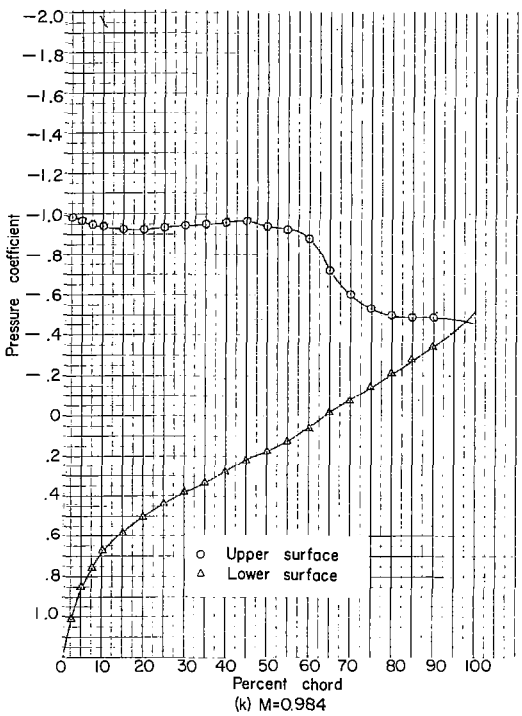
Figure 25.- Continued. NACA 16-009; $\alpha = 12^{\circ}$.



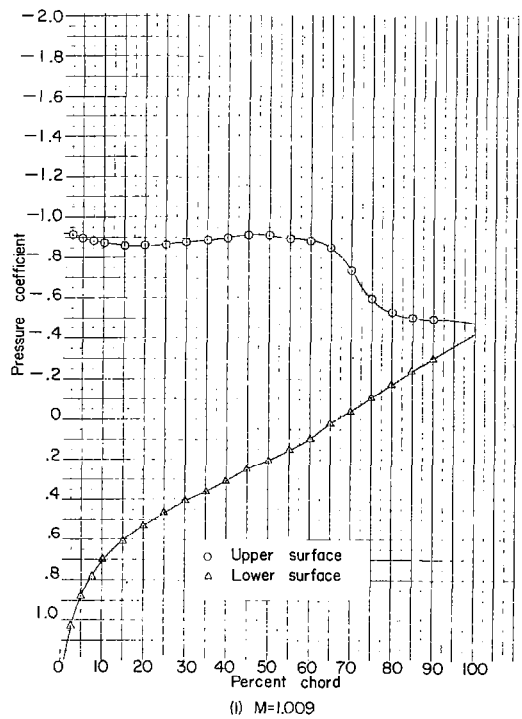
(i) M=0.936



(j) M=0.960

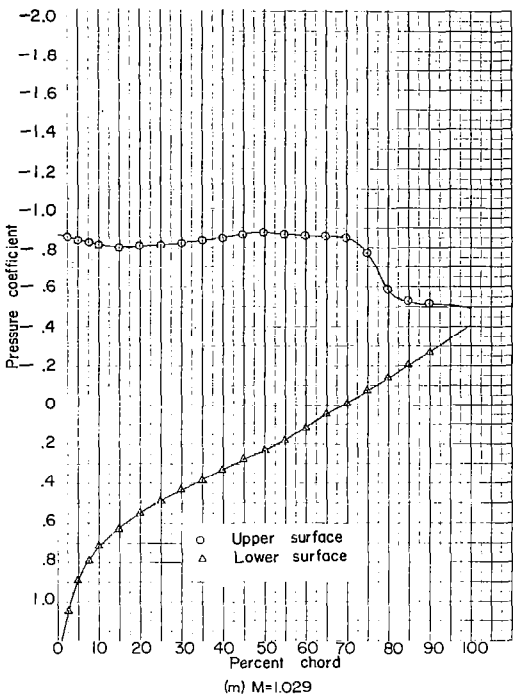


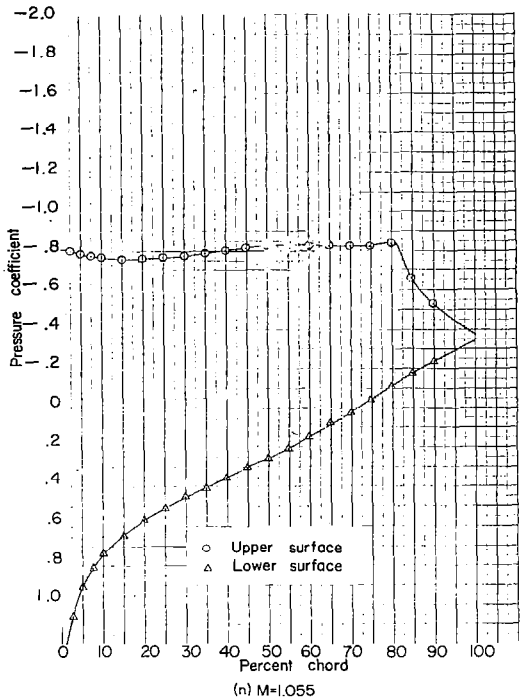
(k) M=0.984

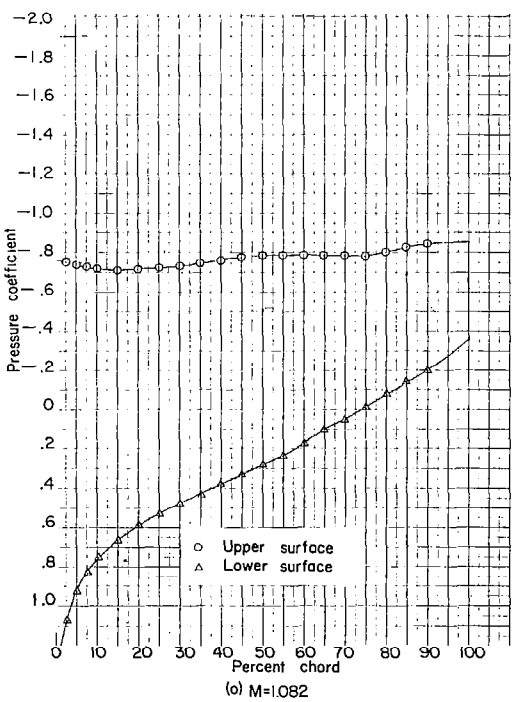


(l) M=1.009

Figure 25.- Continued. NACA 16-009; $\alpha = 12^{\circ}$.







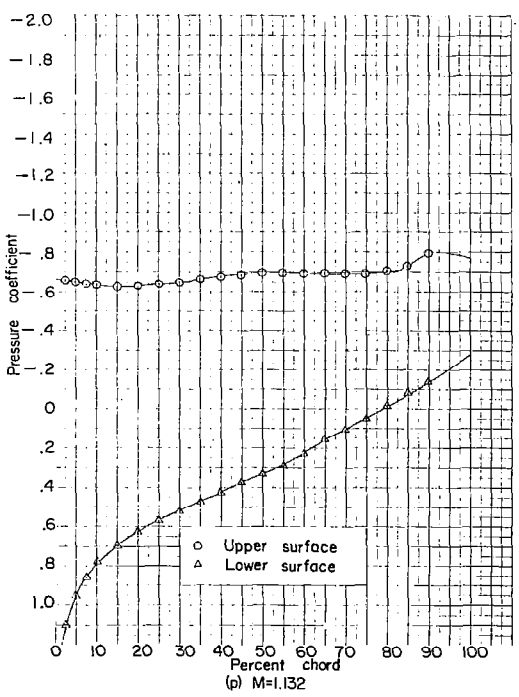


Figure 25.- Concluded. NACA 16-009; $\alpha = 12^{\circ}$.

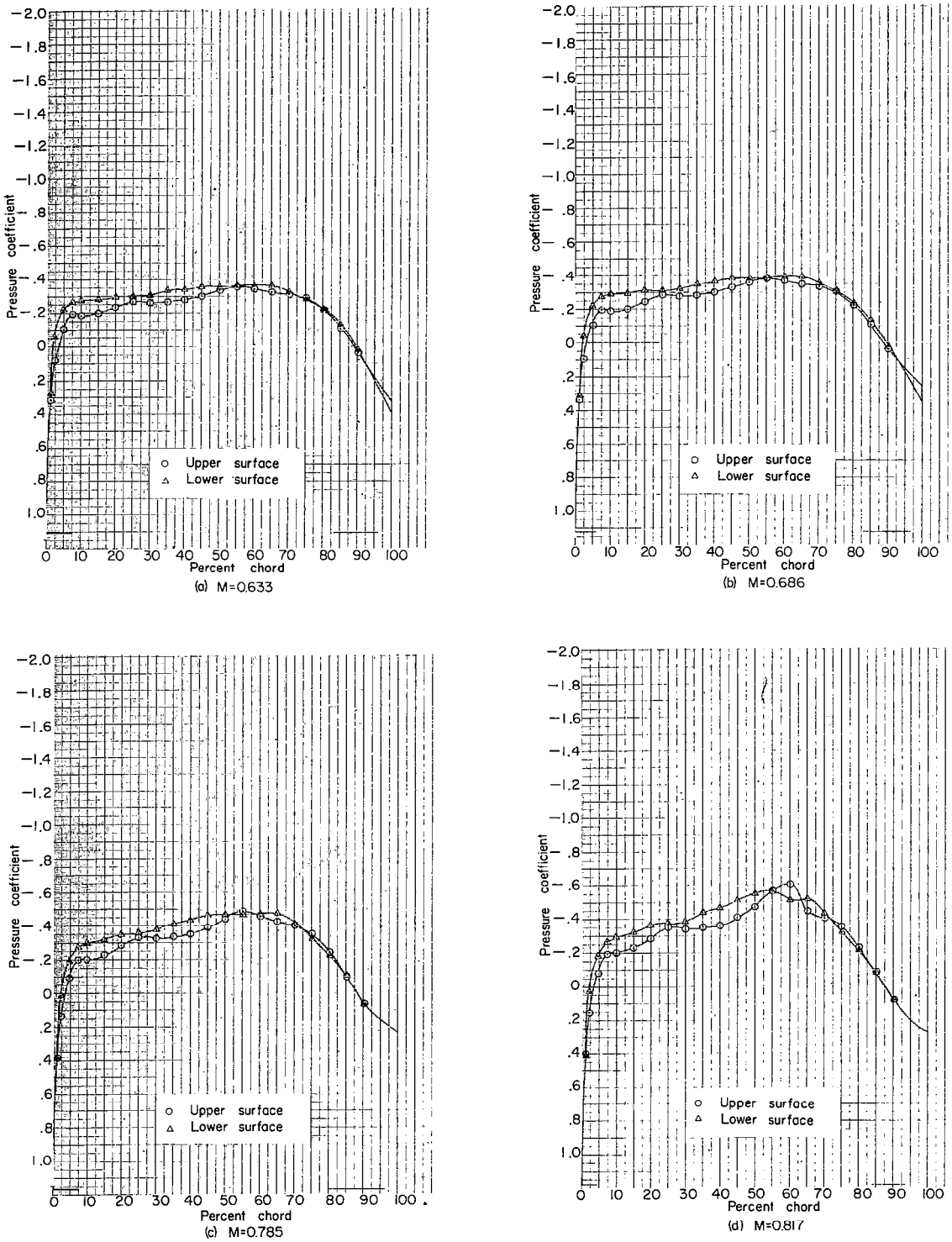
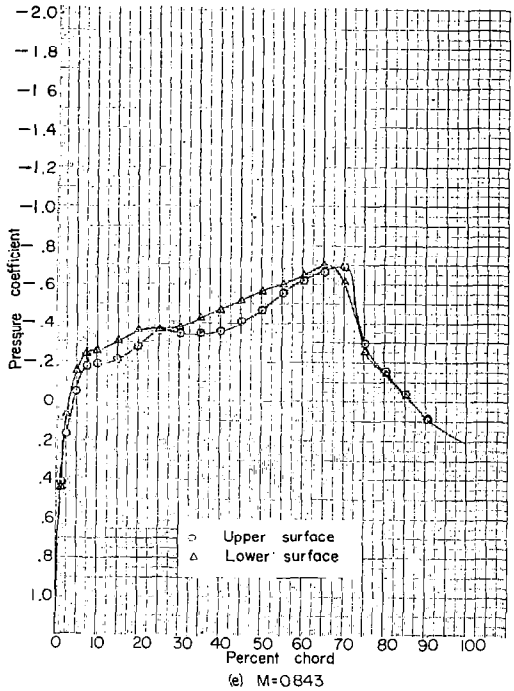
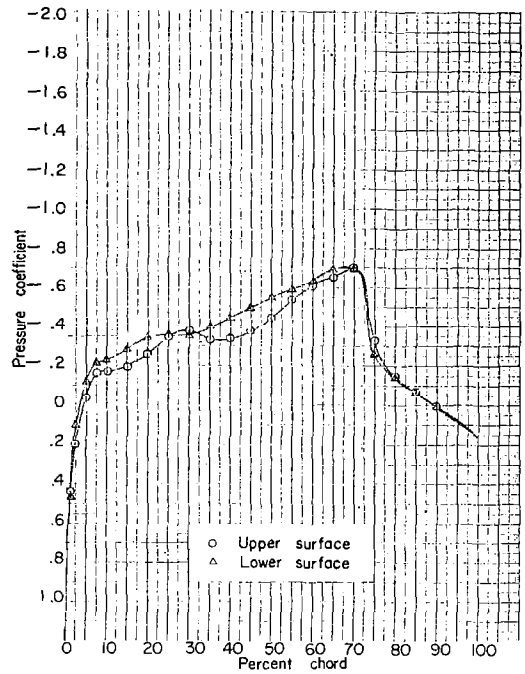


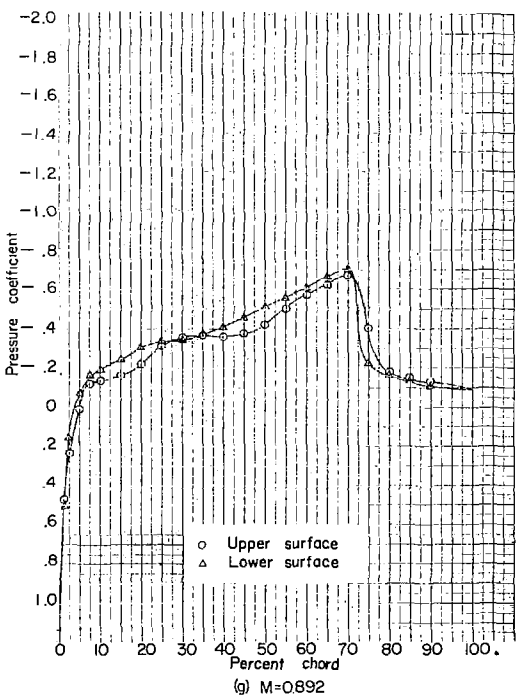
Figure 26.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 0^{\circ}$.



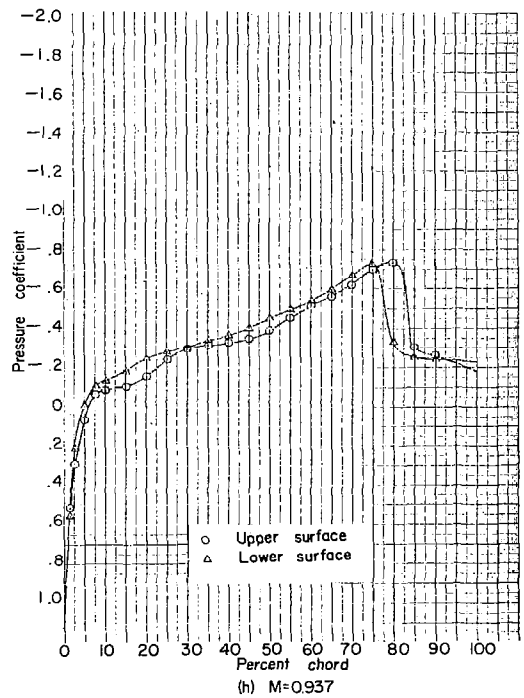
(e) M=0.843



(f) M=0.862

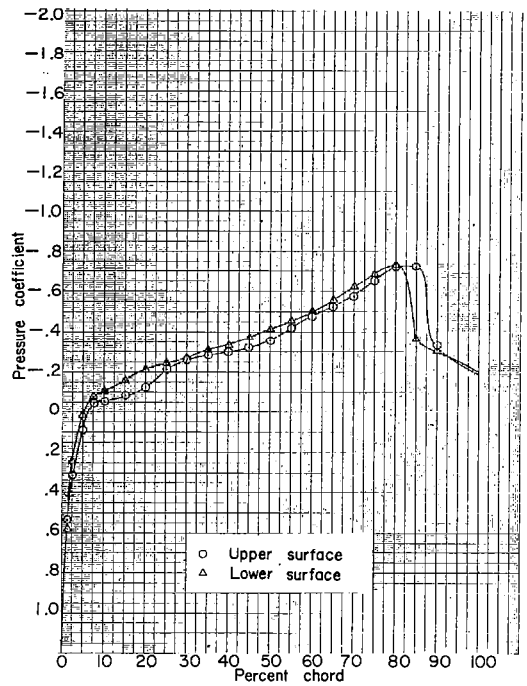


(g) M=0.892

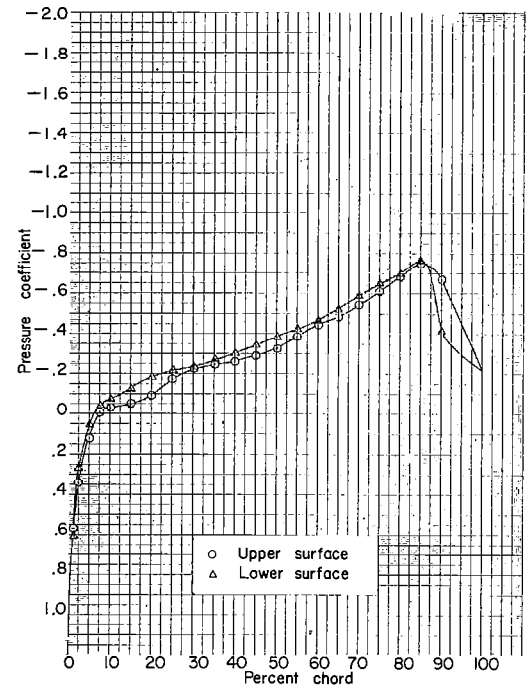


(h) M=0.937

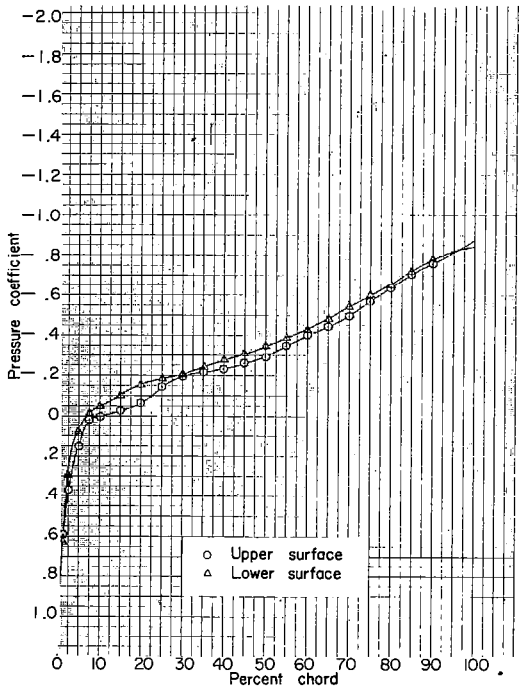
Figure 26.- Continued. NACA 16-012; $\alpha = 0^{\circ}$.



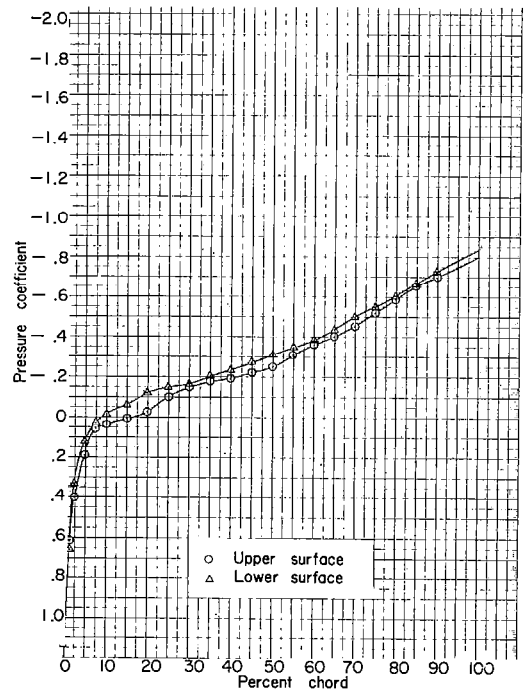
(i) M=0.960



(j) M=0.986

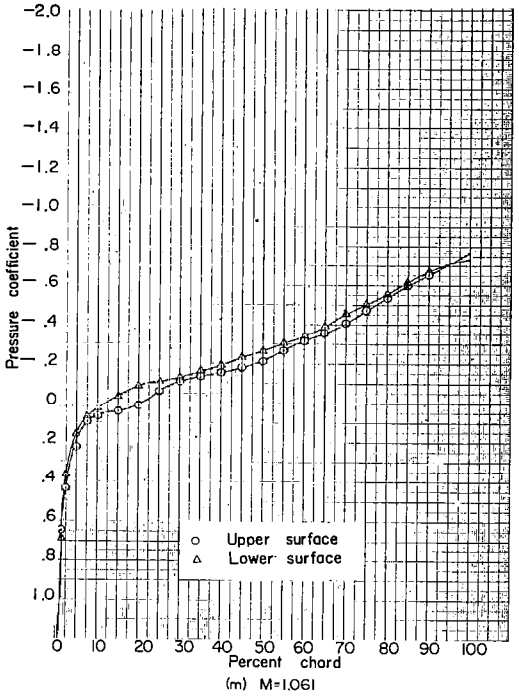


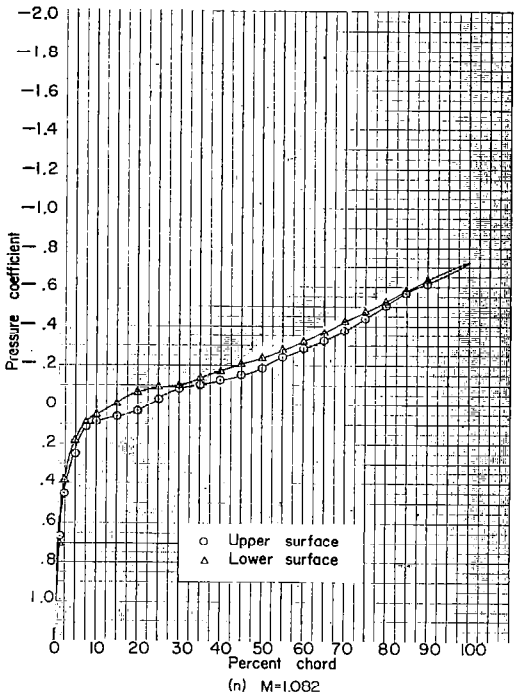
(k) M=1.007

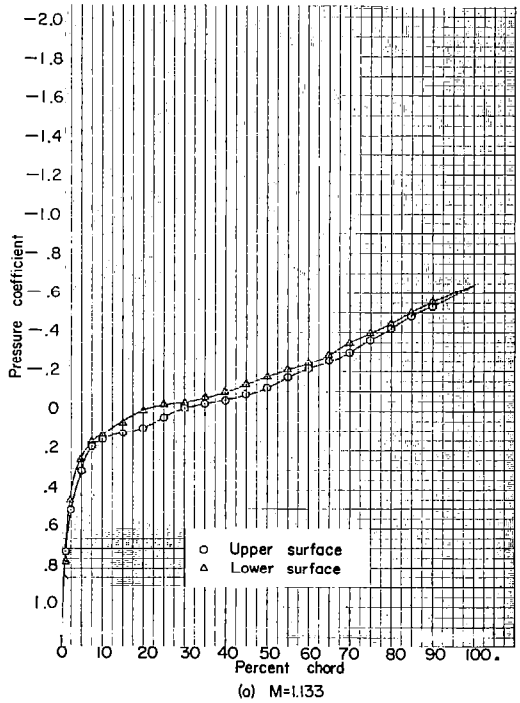


(l) M=1.035

Figure 26.- Continued. NACA 16-012; $\alpha = 0^{\circ}$.







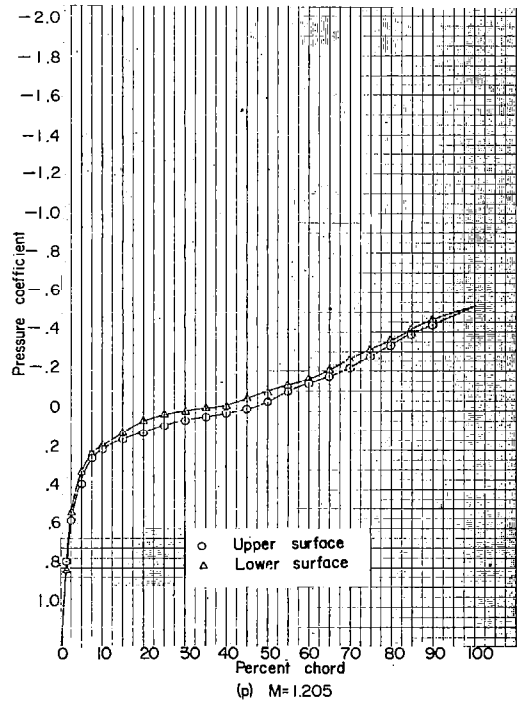


Figure 26.- Concluded. NACA 16-012; $\alpha = 0^{\circ}$.

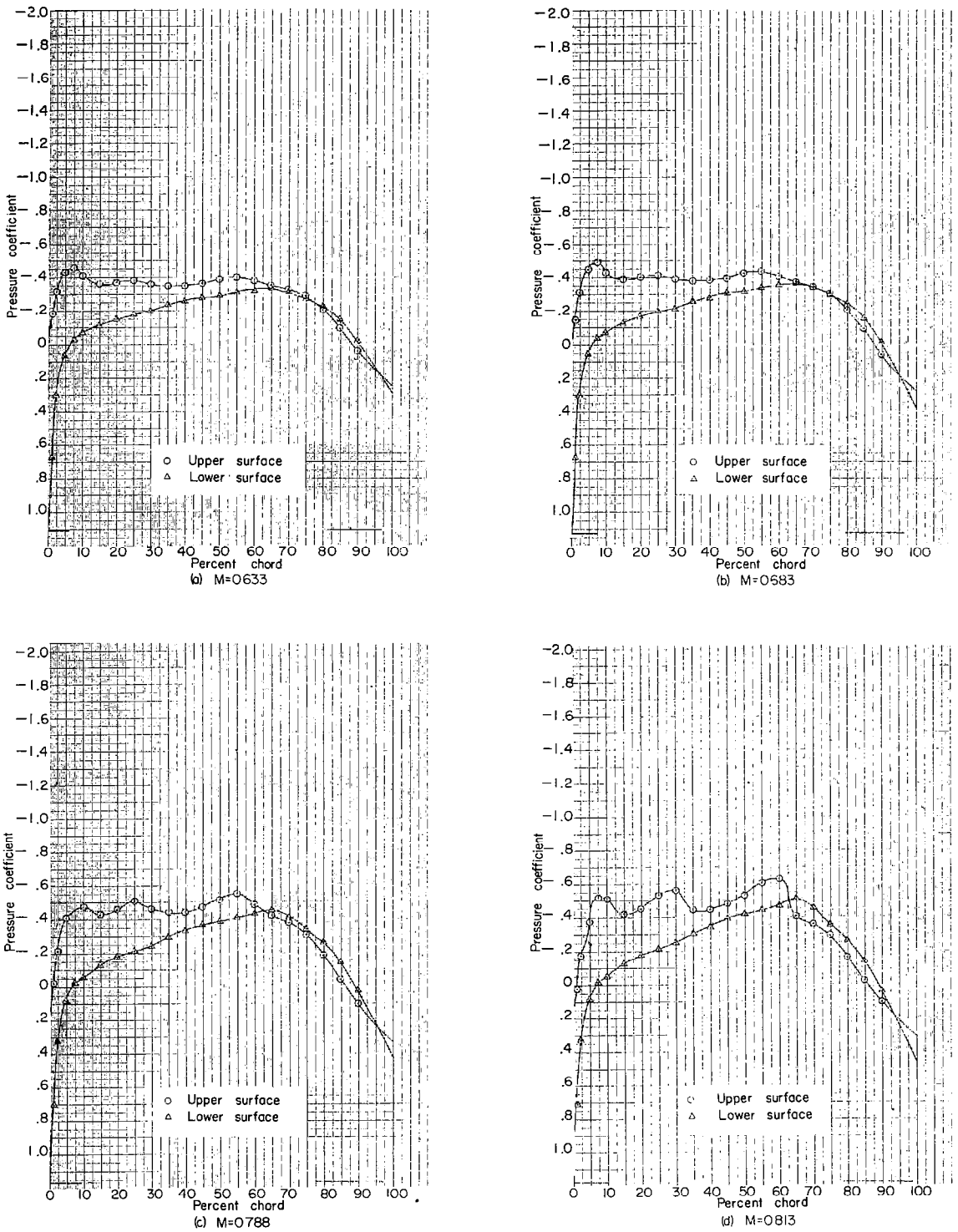
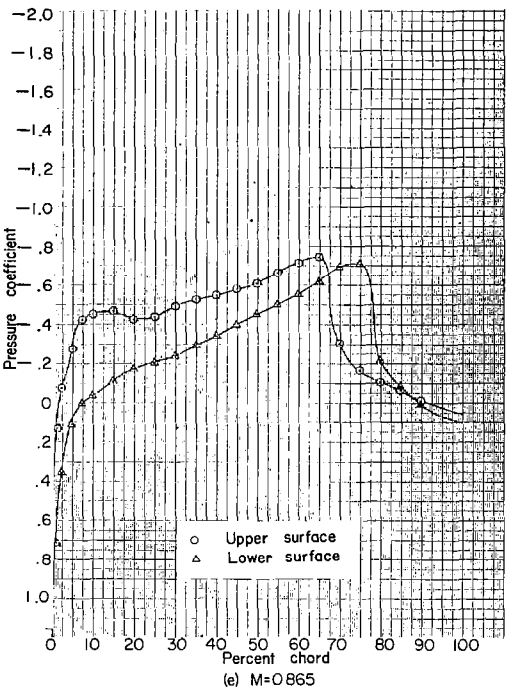
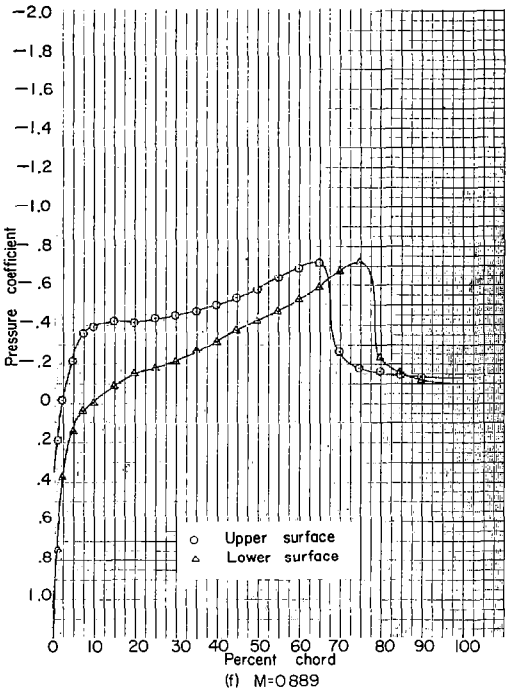


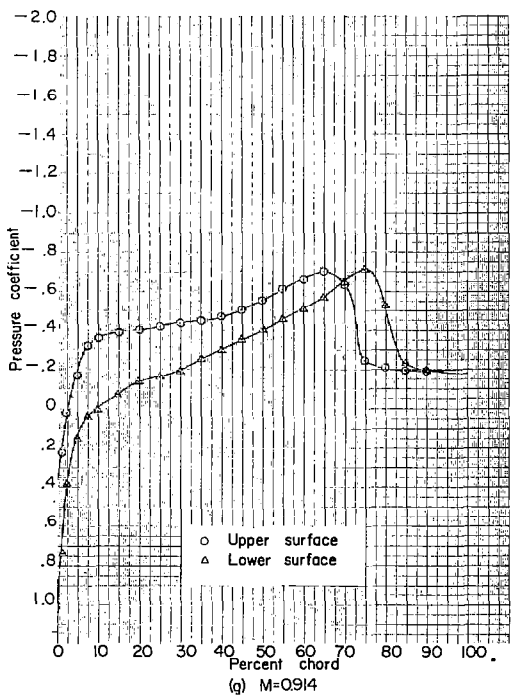
Figure 27.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 2^{\circ}$.



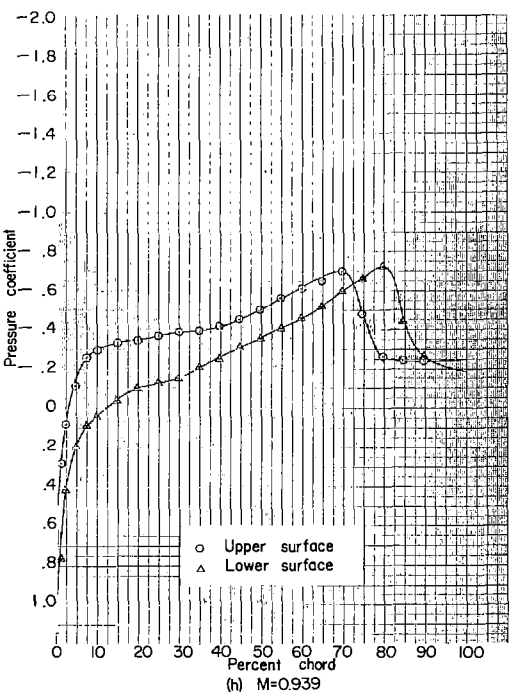
(e) M=0.865



(f) M=0.889

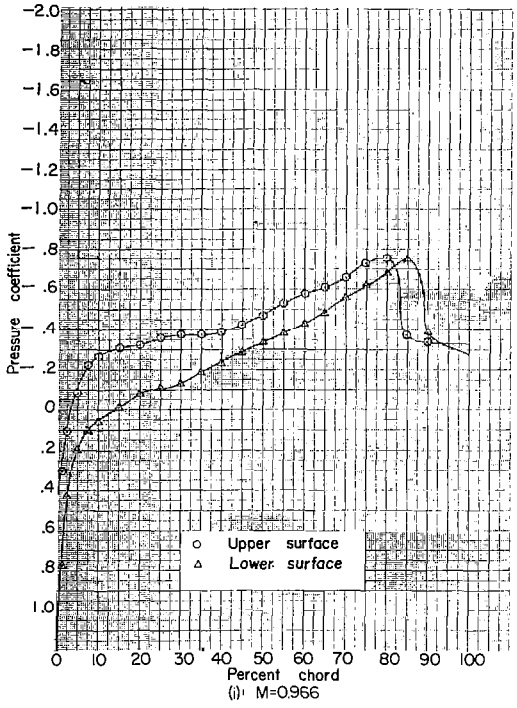


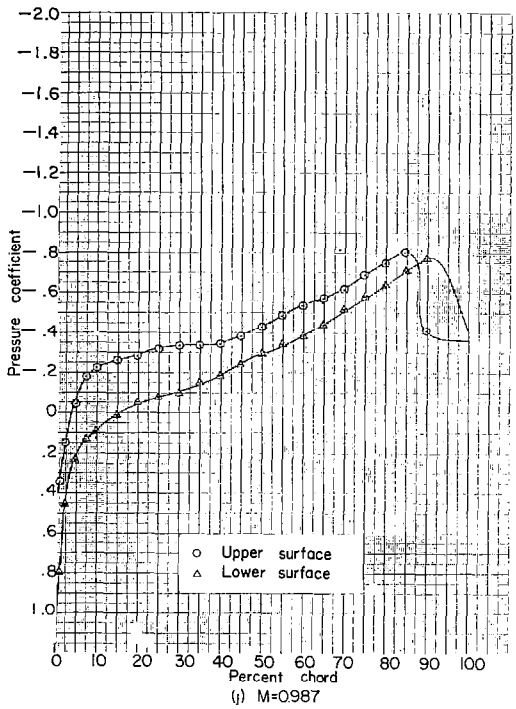
(g) M=0.914

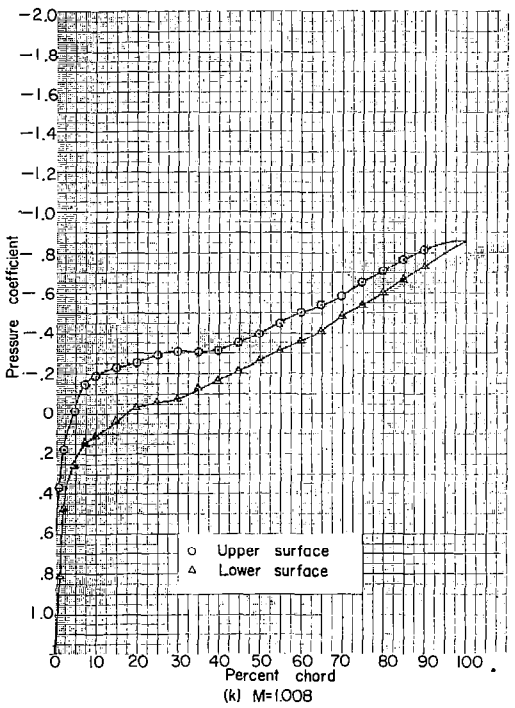


(h) M=0.939

Figure 27.- Continued. NACA 16-012; $\alpha = 2^{\circ}$.







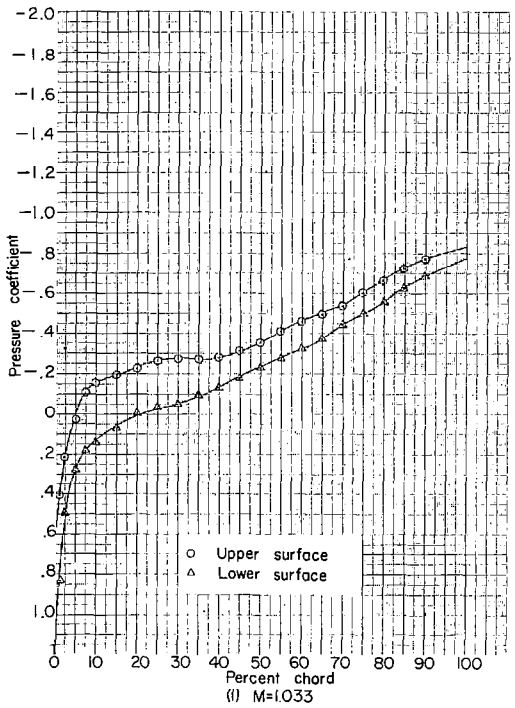
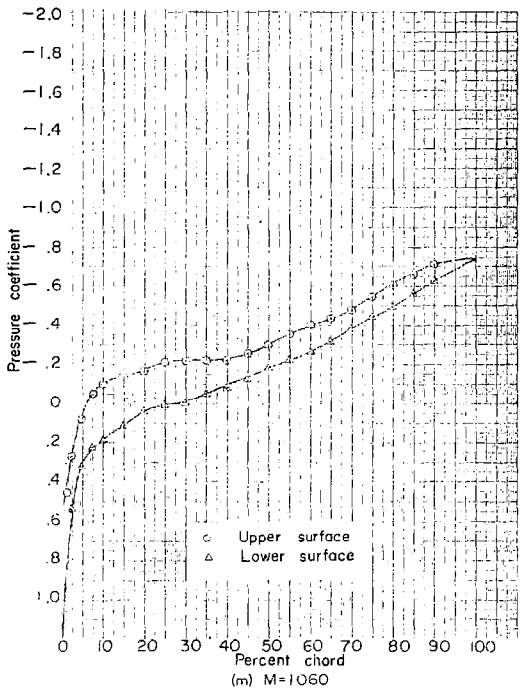
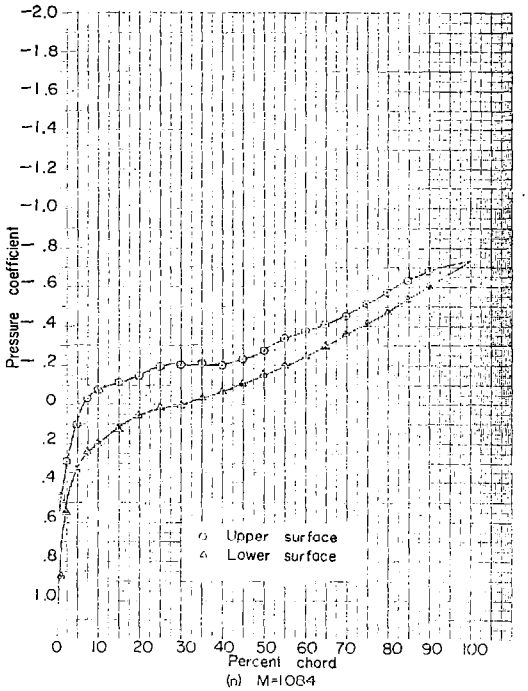
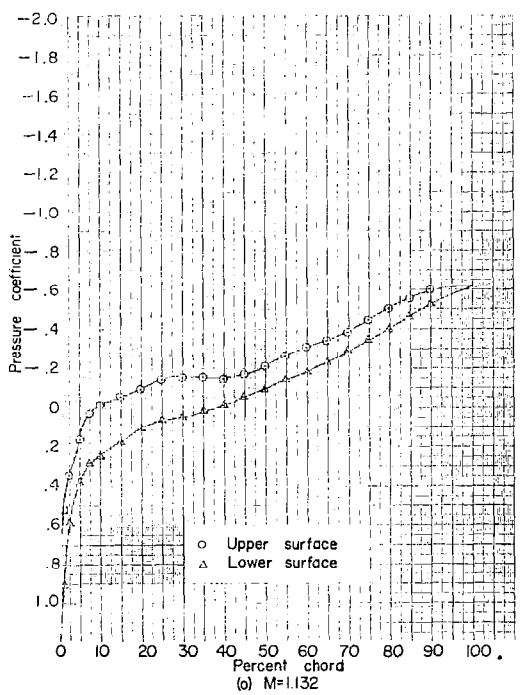


Figure 27.- Continued. NACA 16-012; $\alpha = 2^{\circ}$.







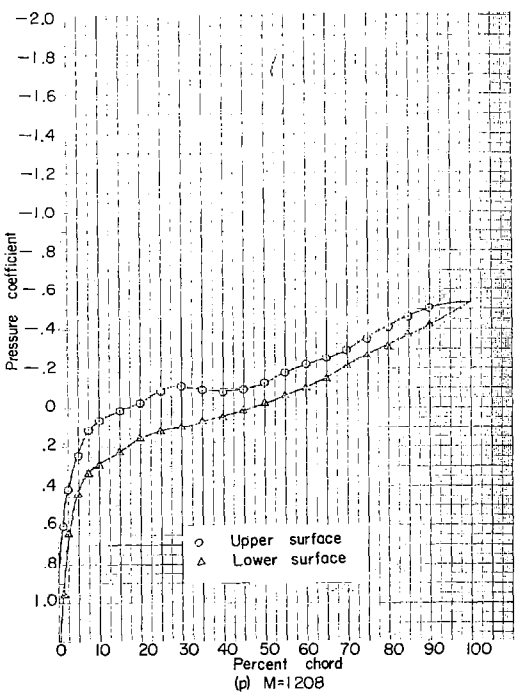


Figure 27.- Concluded. NACA 16-012; $\alpha = 2^{o}$.

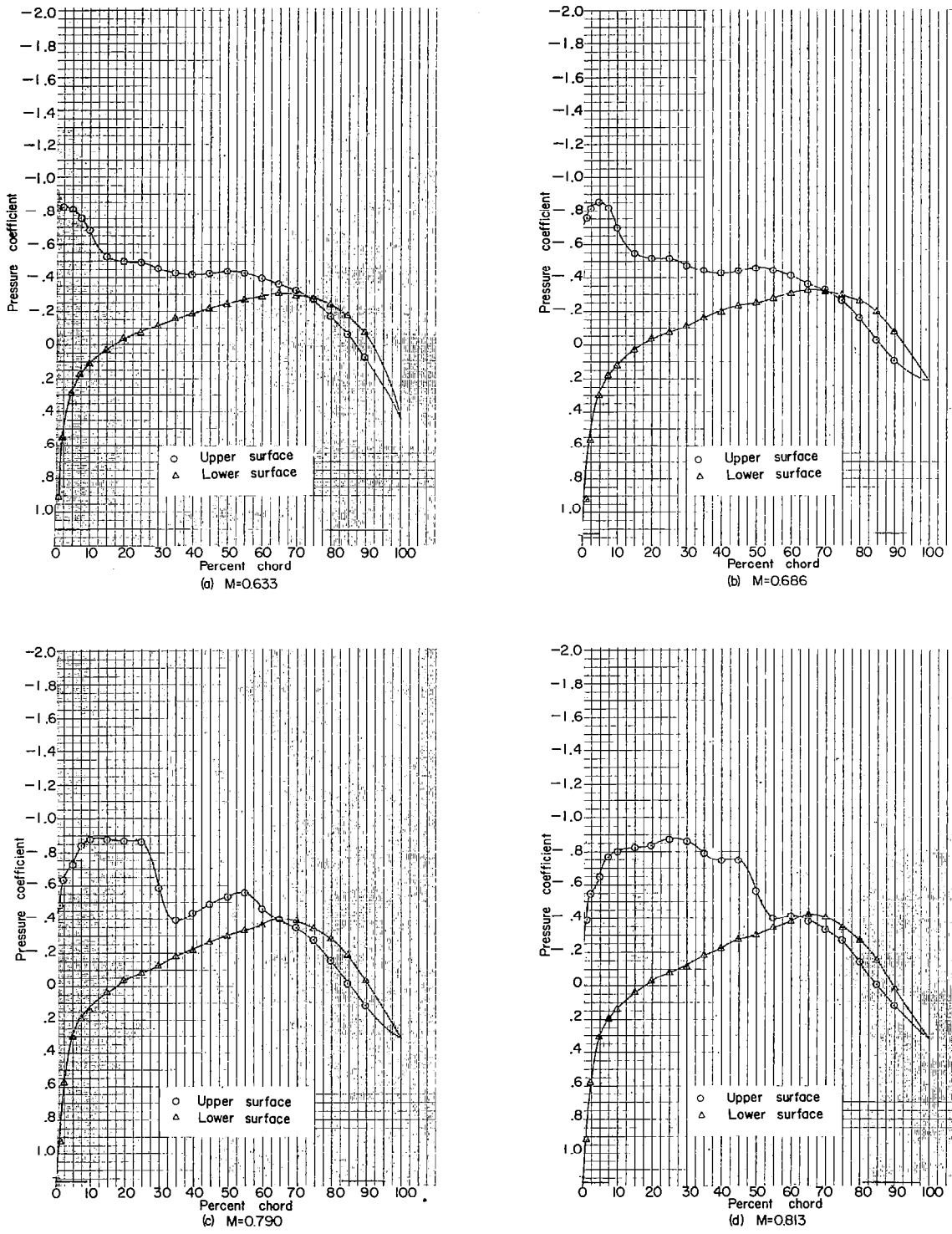
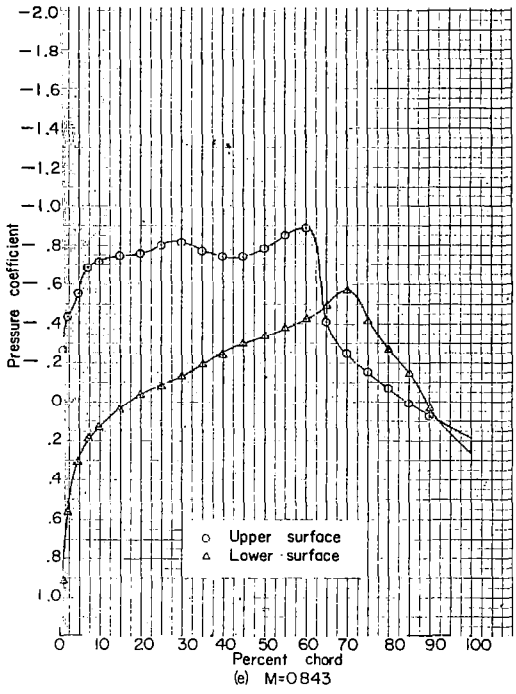
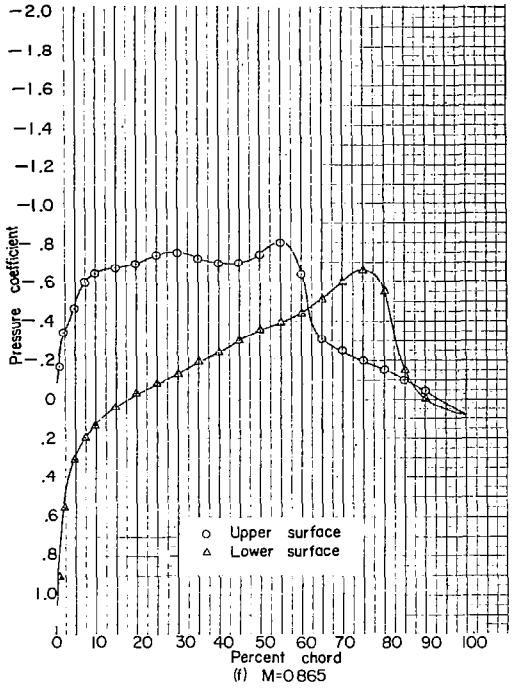
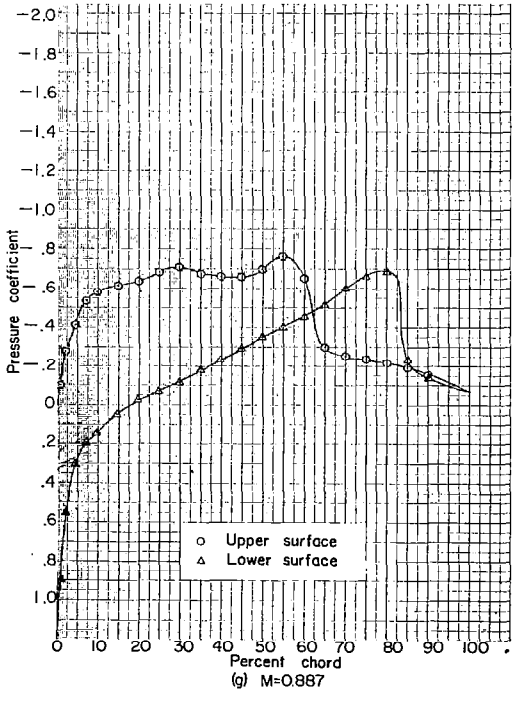


Figure 28.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 4^{\circ}$.







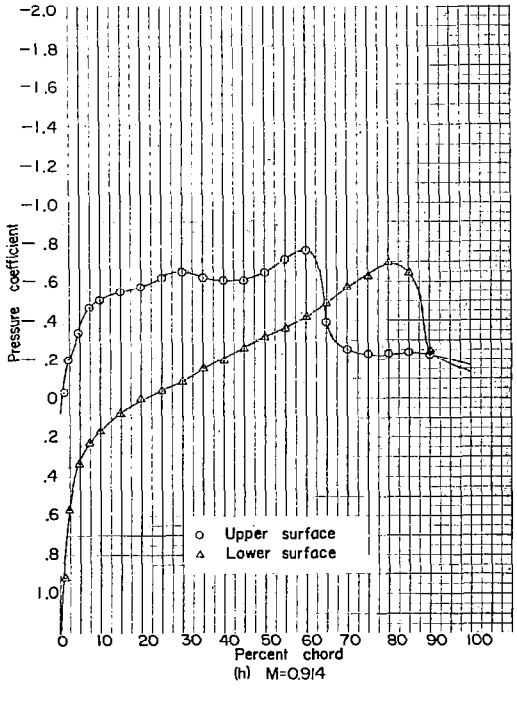
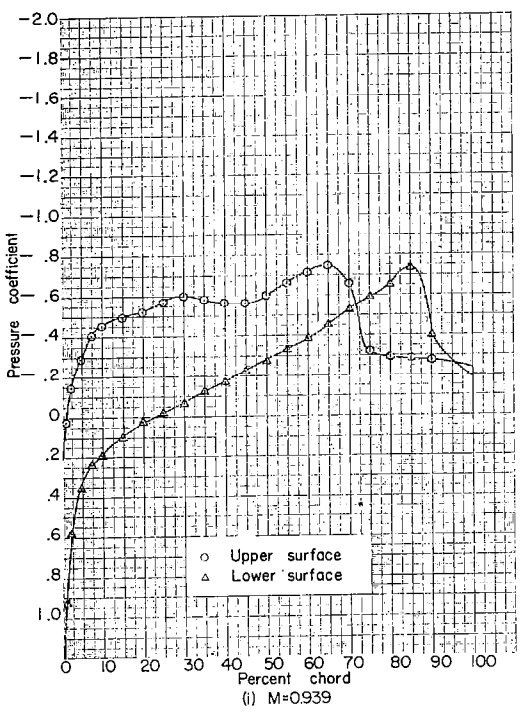
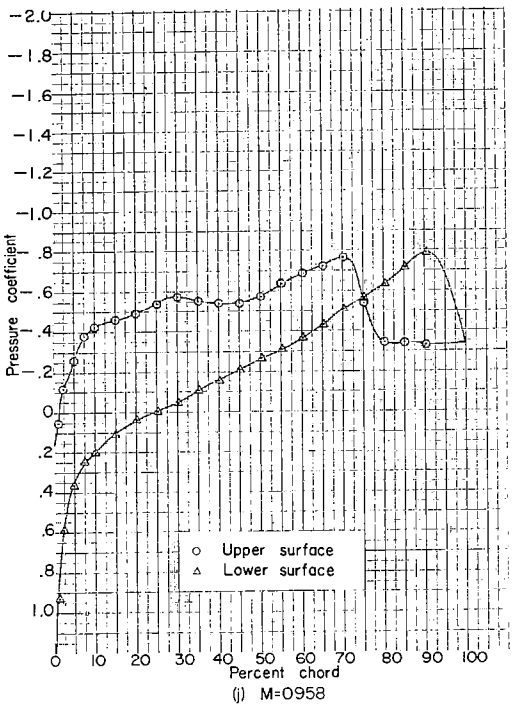
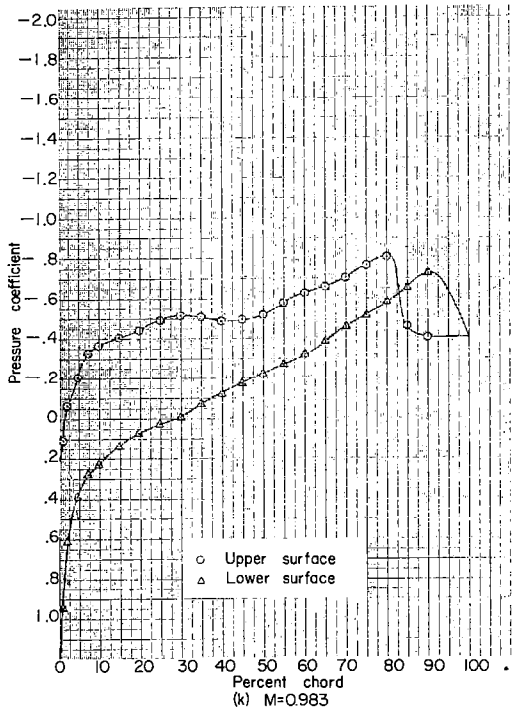


Figure 28.- Continued. NACA 16-012; $\alpha = 4^{\circ}$.







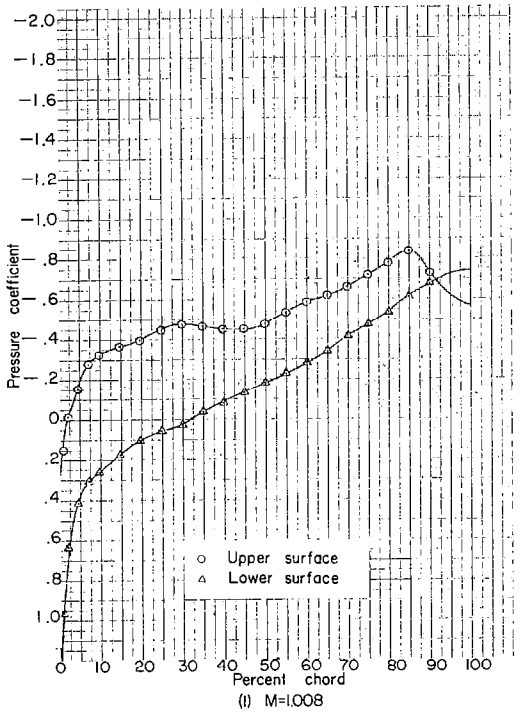
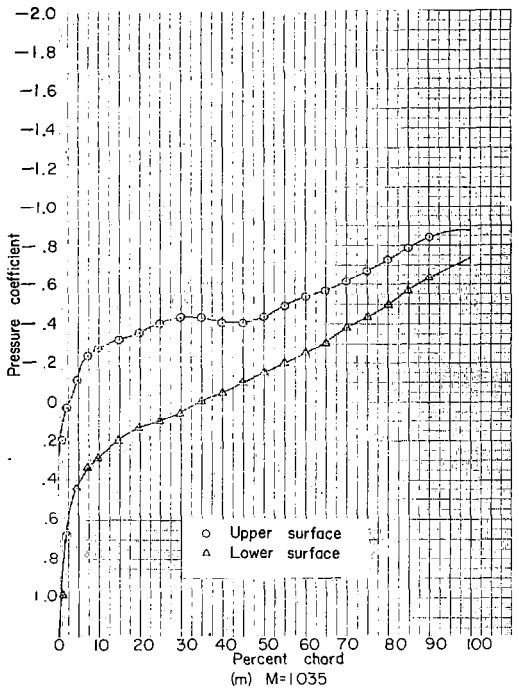
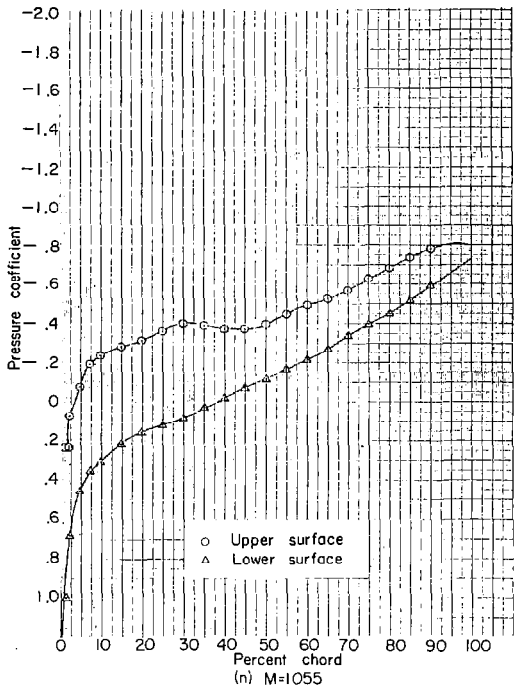
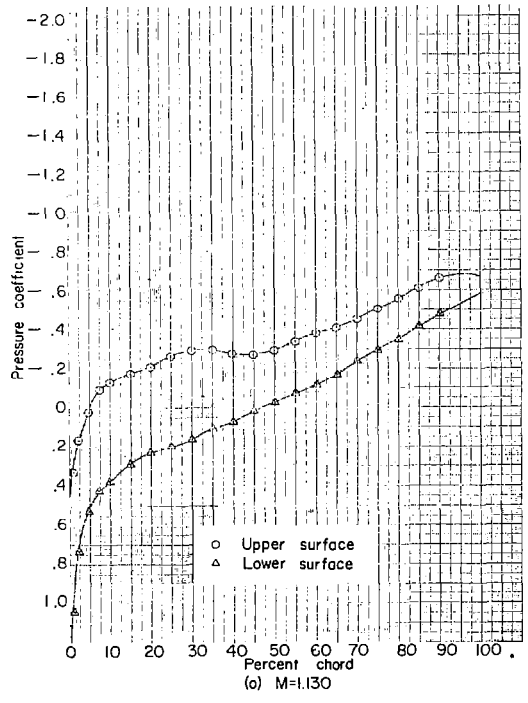


Figure 28.- Continued. NACA 16-012; $\alpha = 4^{o}$.







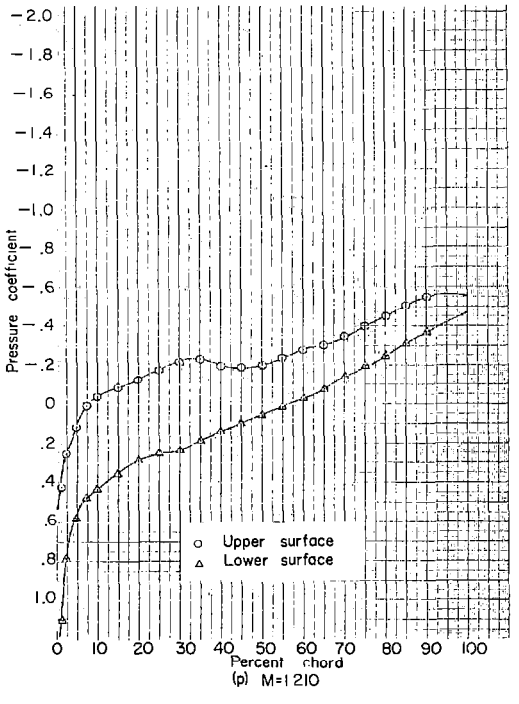


Figure 28.- Concluded. NACA 16-012; $\alpha = 4^{\circ}$.

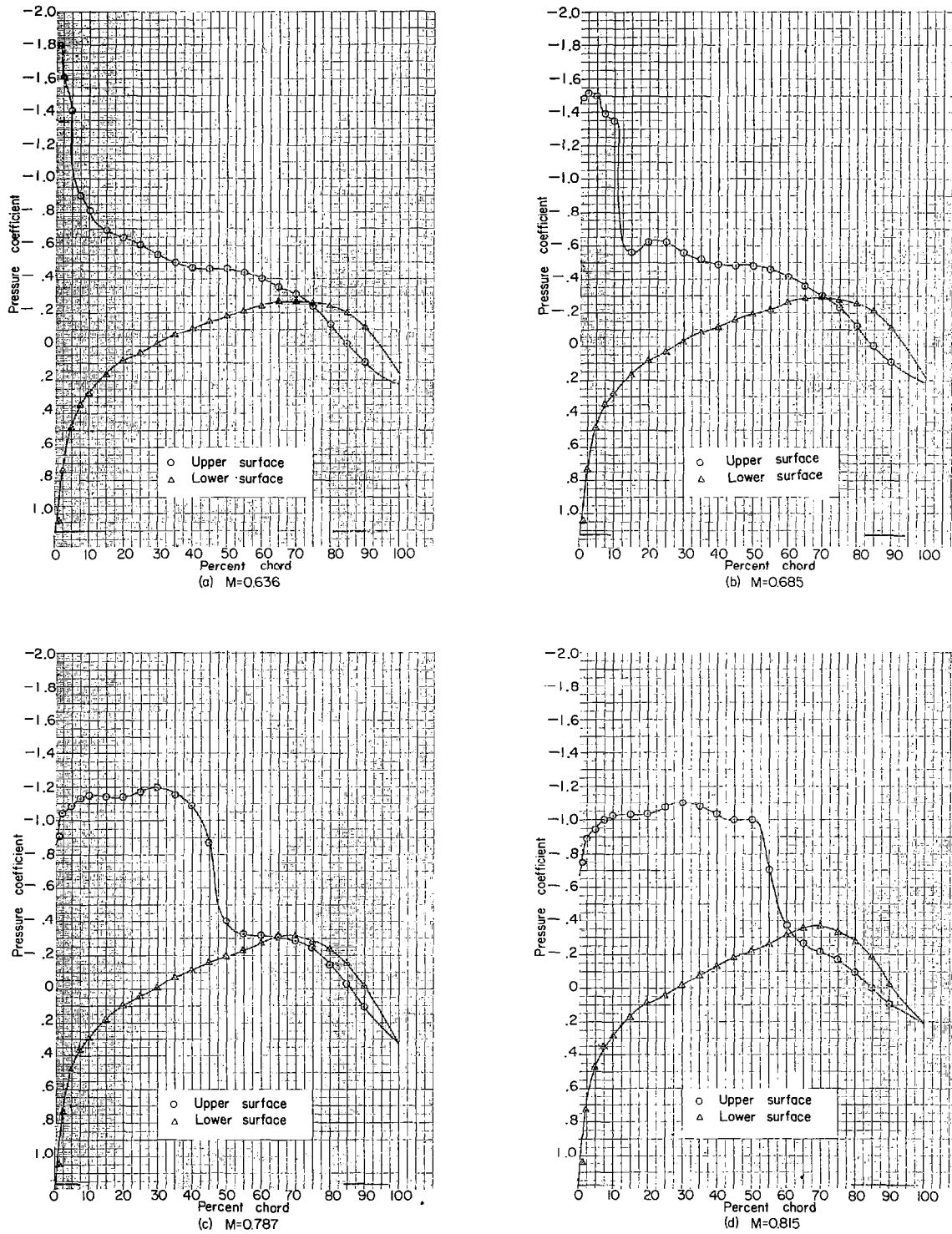
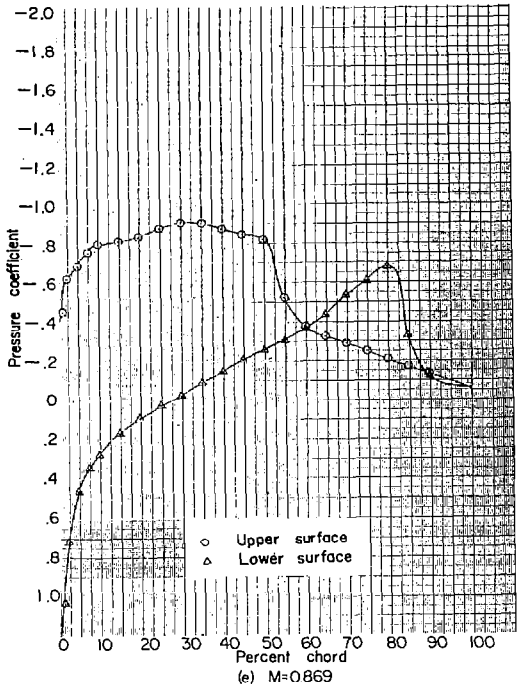
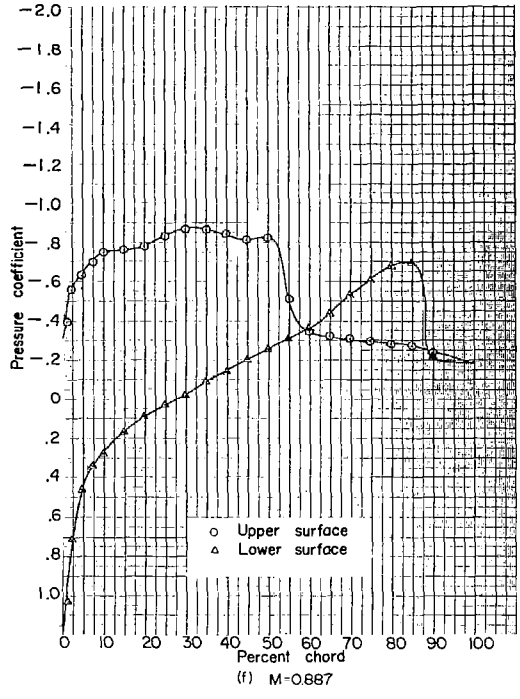
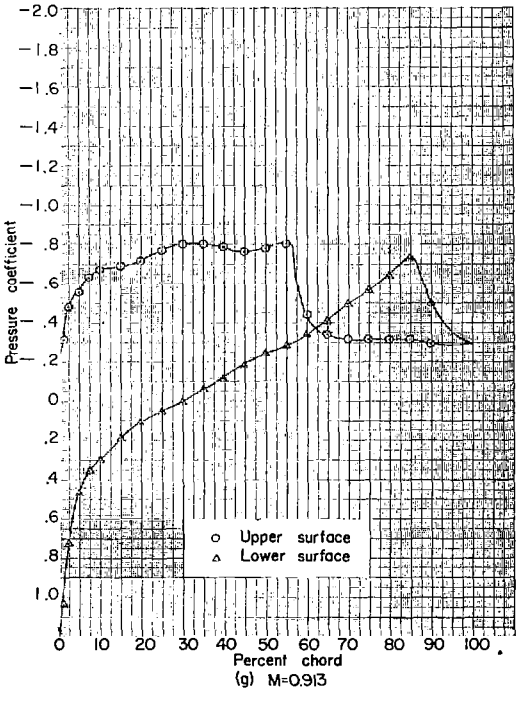


Figure 29.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 6^{\circ}$.







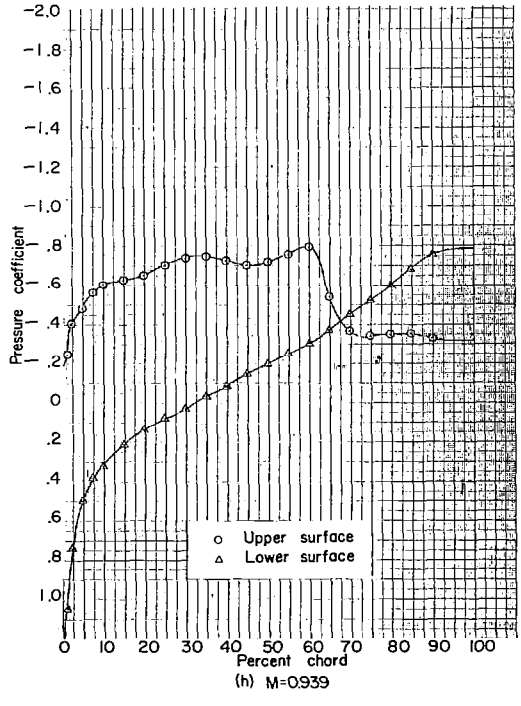
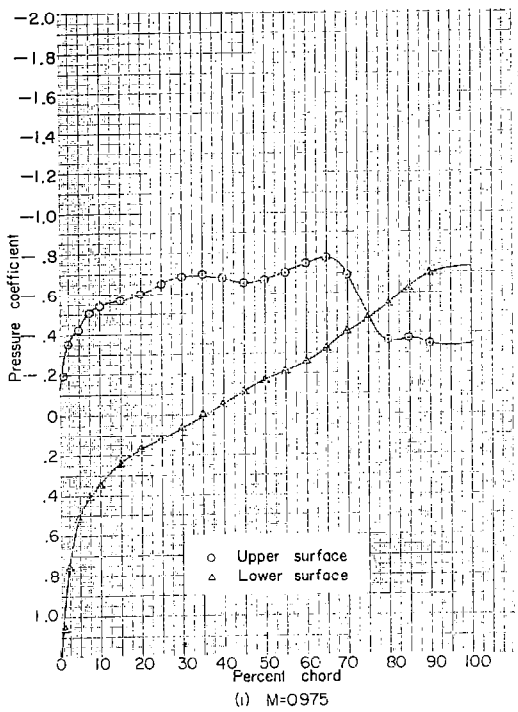
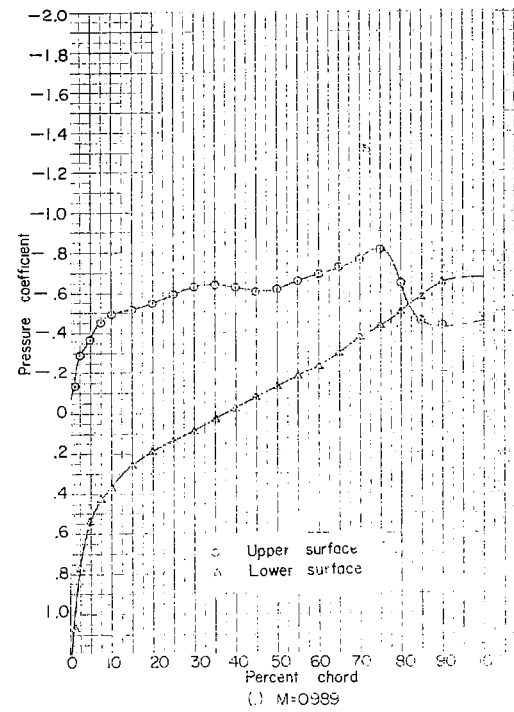
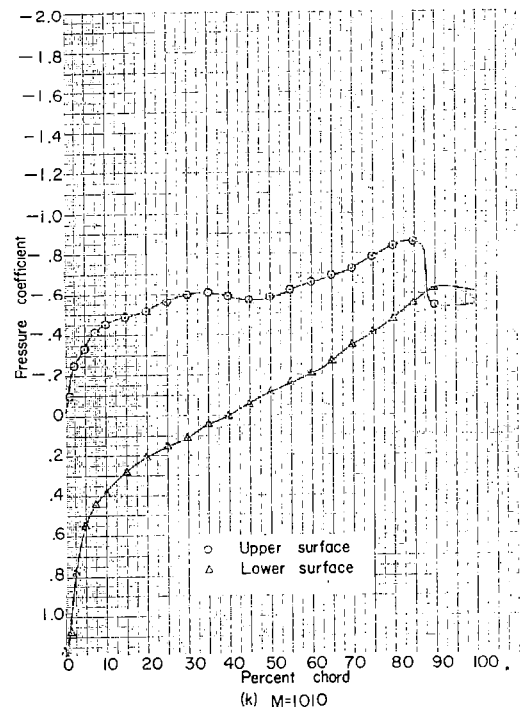


Figure 29.- Continued. NACA 16-012; $\alpha = 6^{\circ}$.







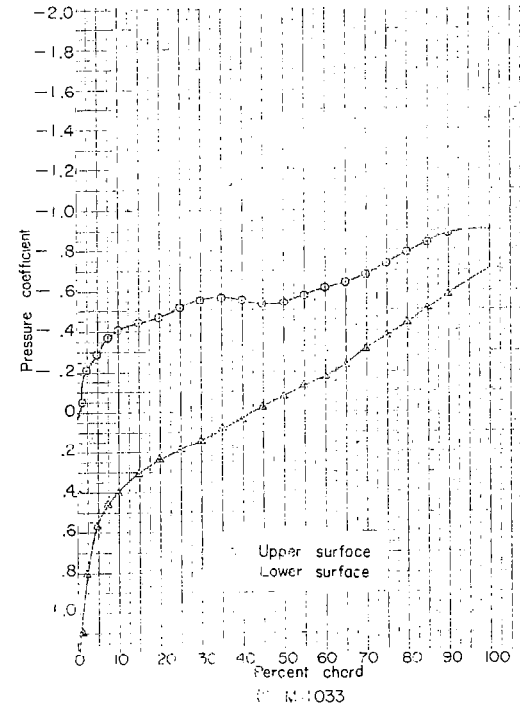
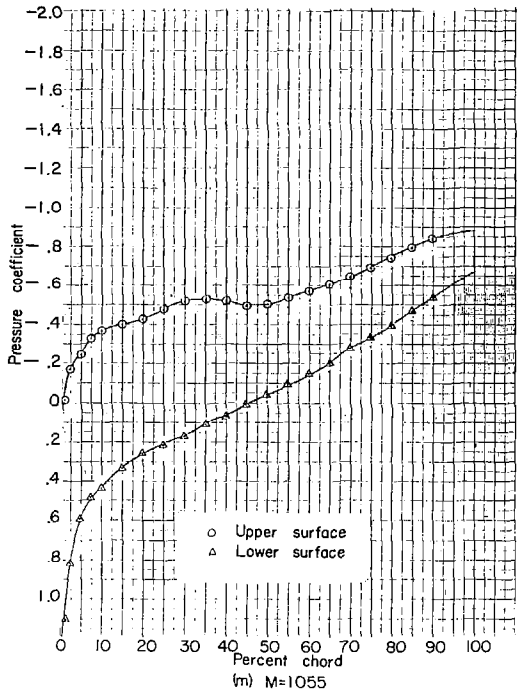
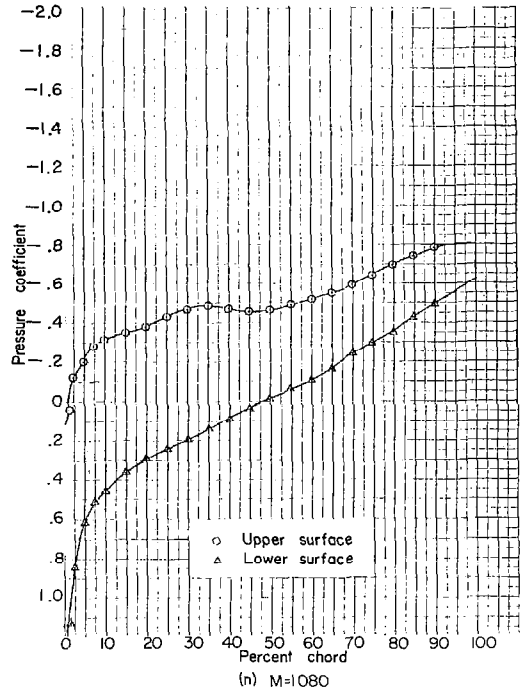
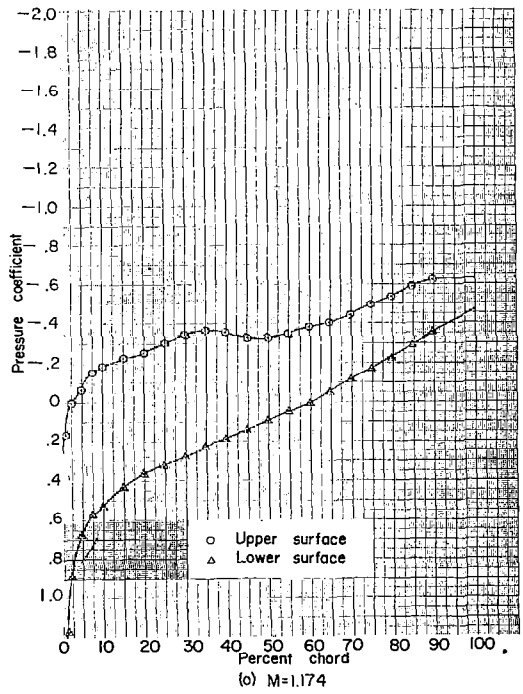


Figure 29.- Continued. NACA 16-012; $\alpha = 6^{o}$.







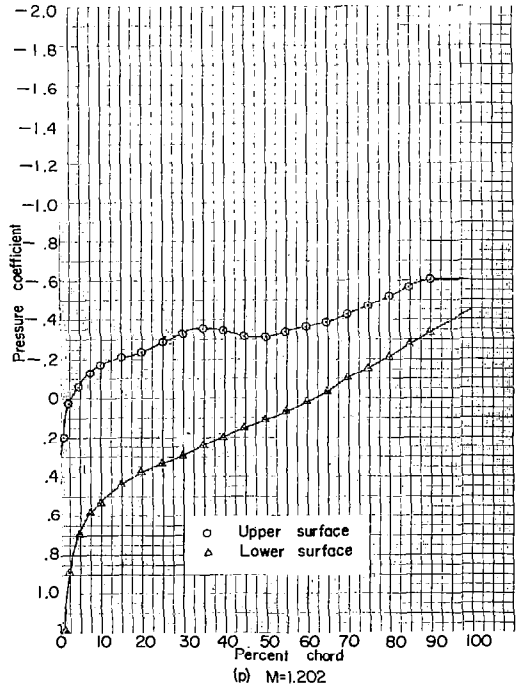


Figure 29.- Concluded. NACA 16-012; $\alpha = 6^{\circ}$.

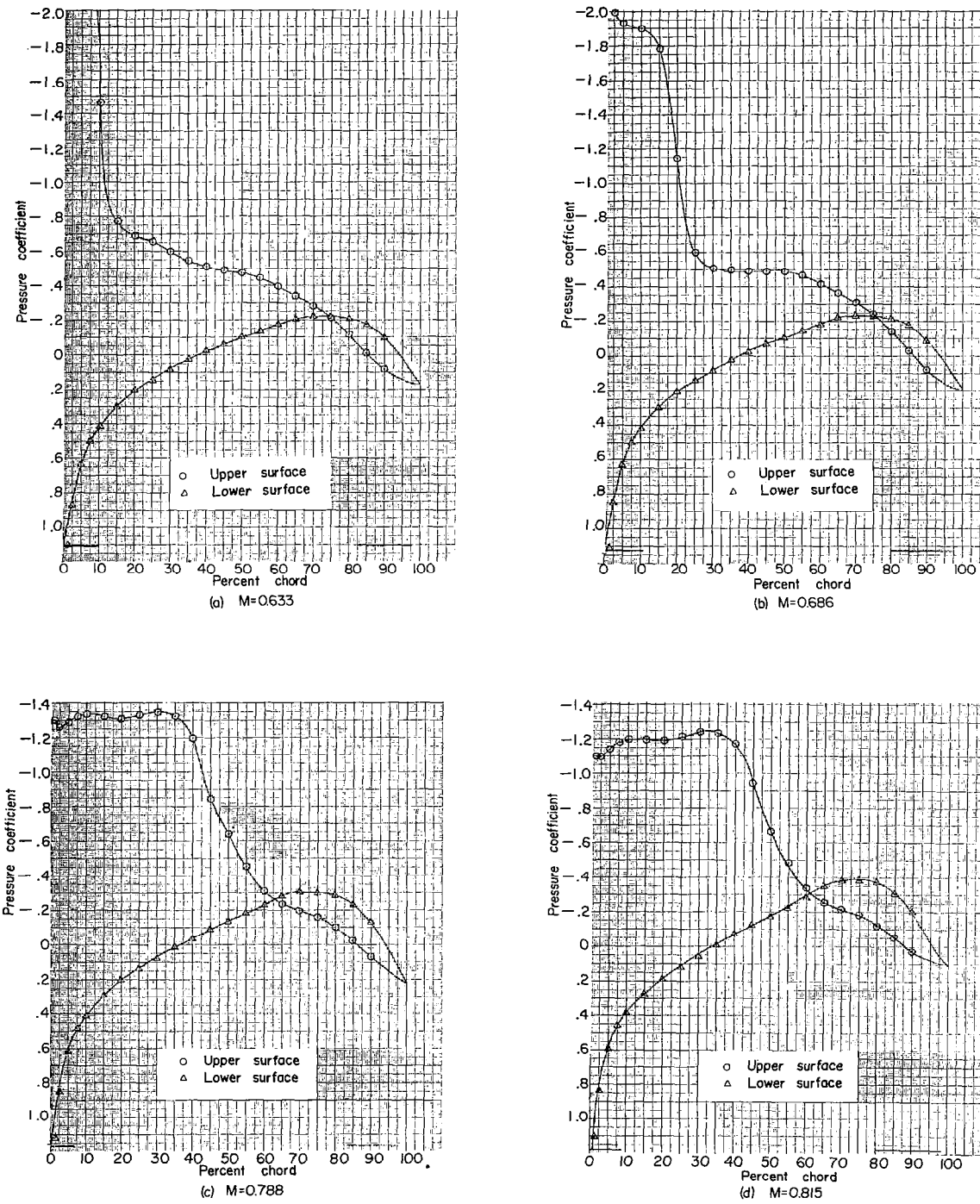
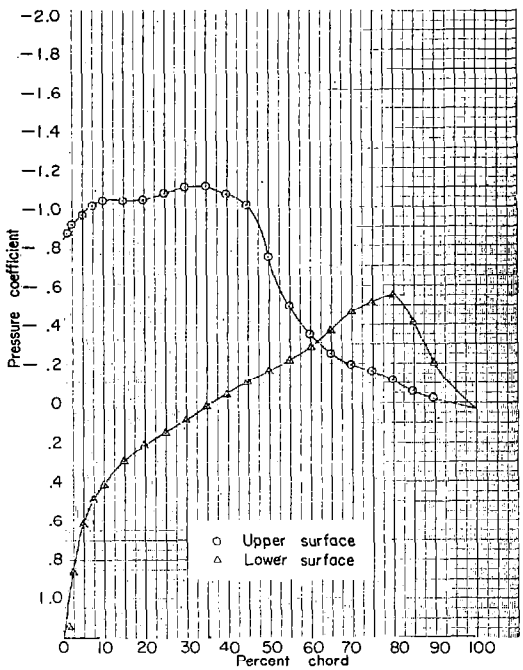
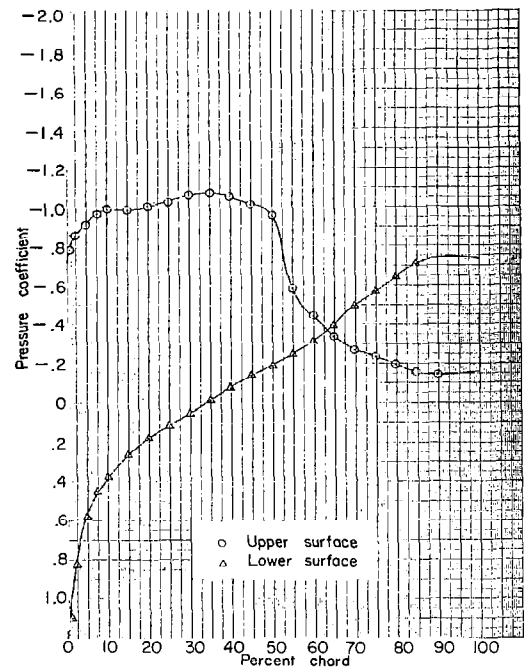


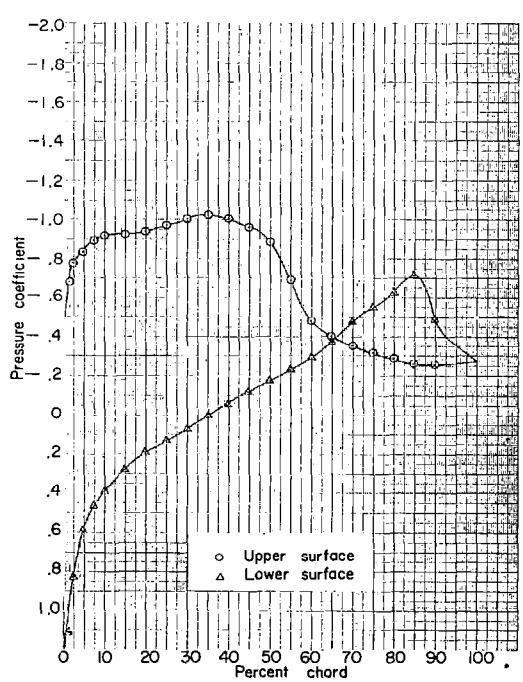
Figure 30.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 8^{\circ}$.



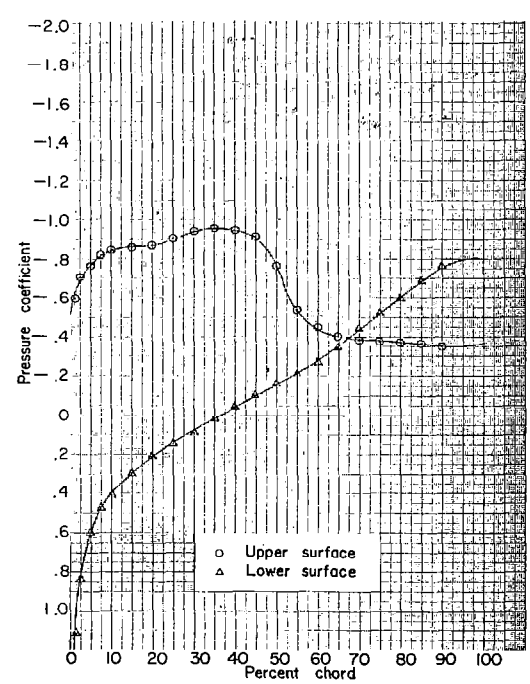
(e) M=0.846



(f) M=0.867

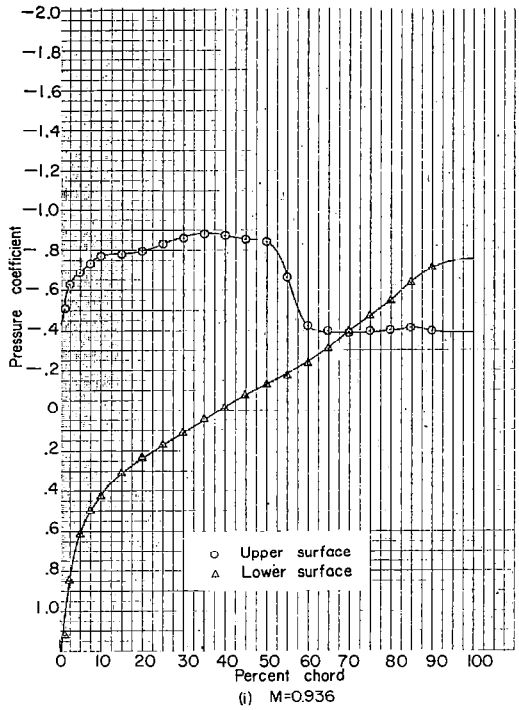


(g) M=0.887

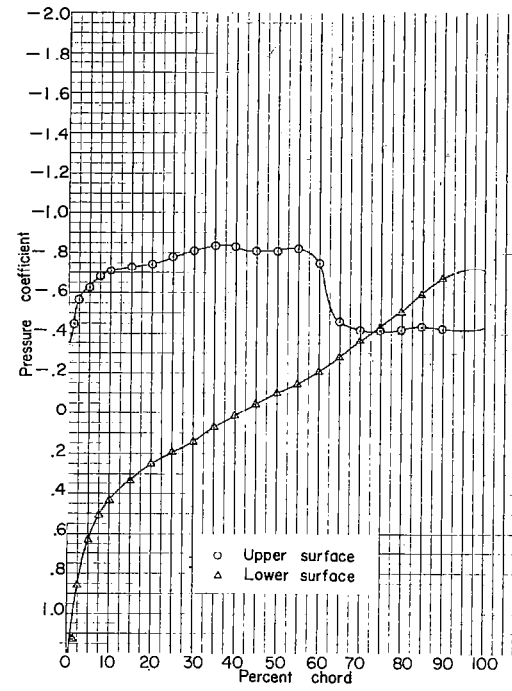


(h) M=0.907

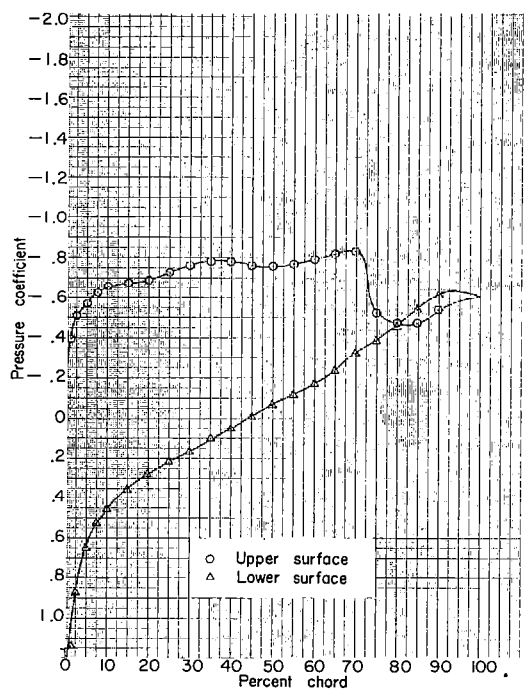
Figure 30.- Continued. NACA 16-012; $\alpha = 8^{\circ}$.



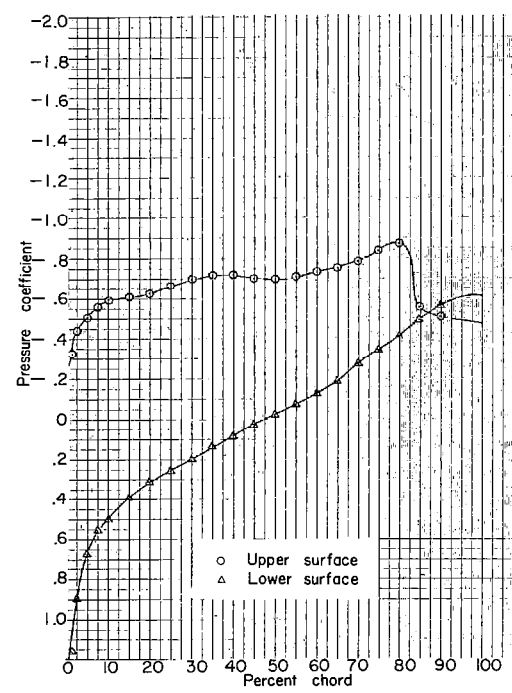
(i) M=0.936



(j) M=0.960

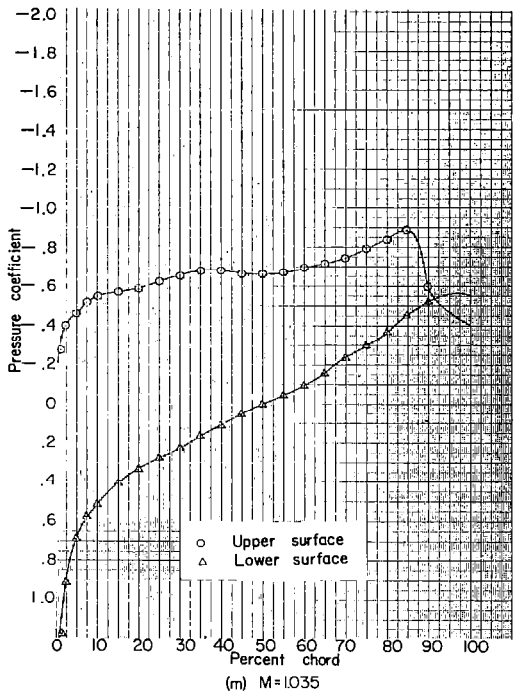


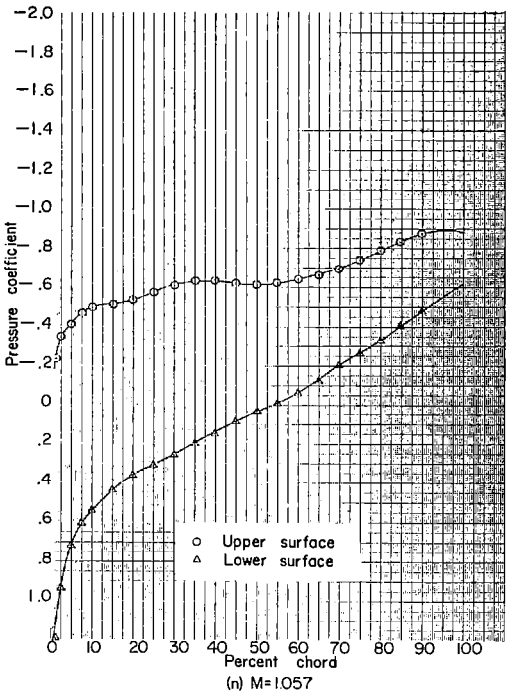
(k) M=0.984

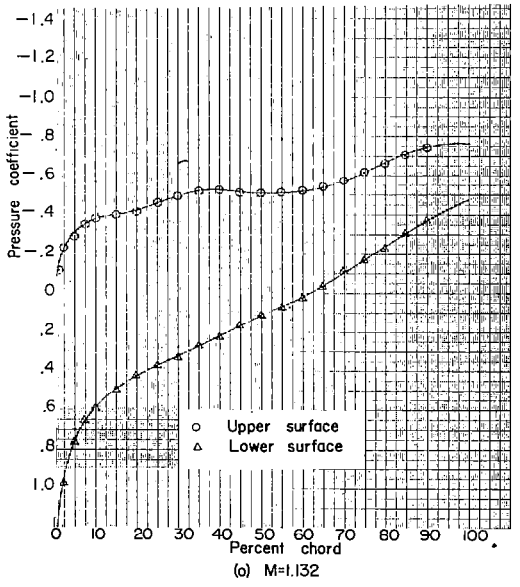


(l) M=1.008

Figure 30.- Continued. NACA 16-012; $\alpha = 8^{\circ}$.







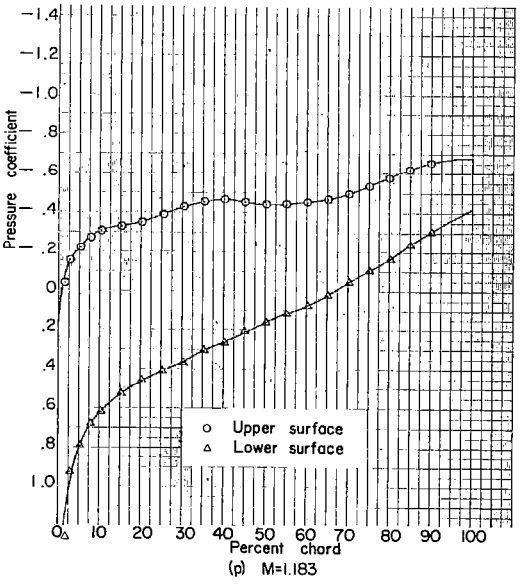


Figure 30.- Concluded. NACA 16-012; $\alpha = 8^{\circ}$.

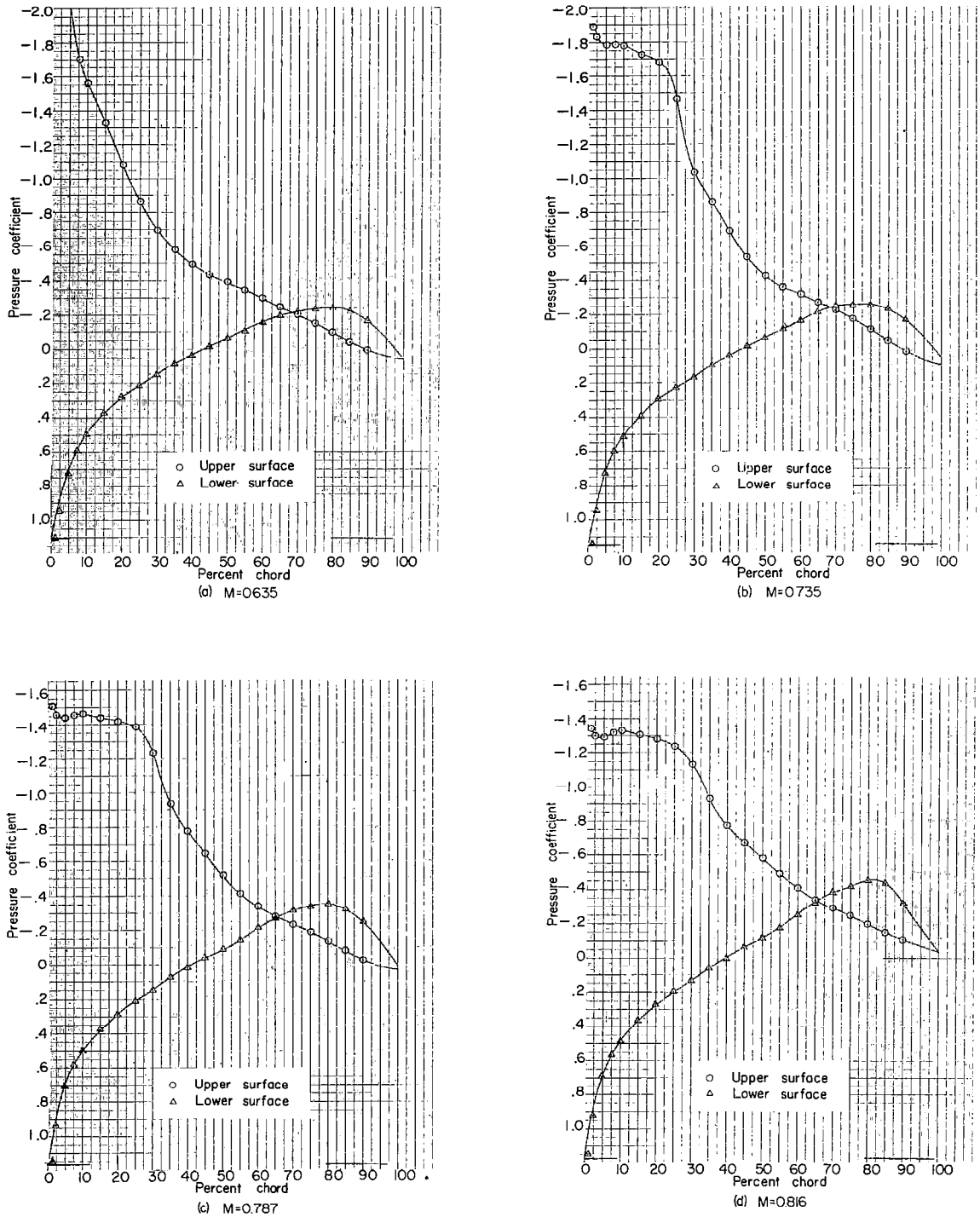
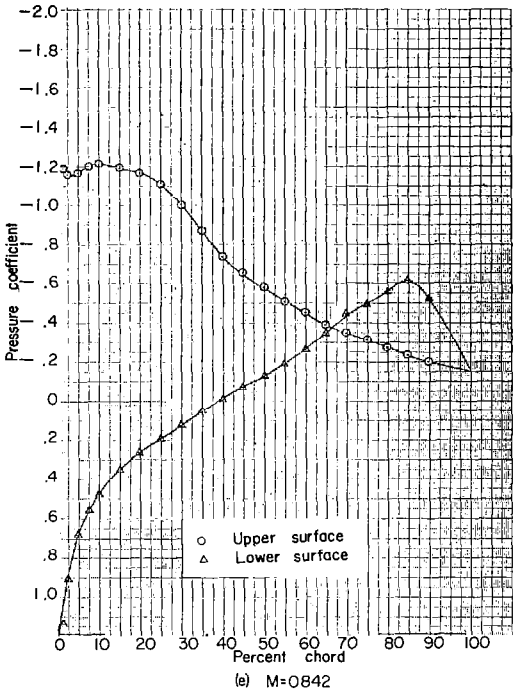
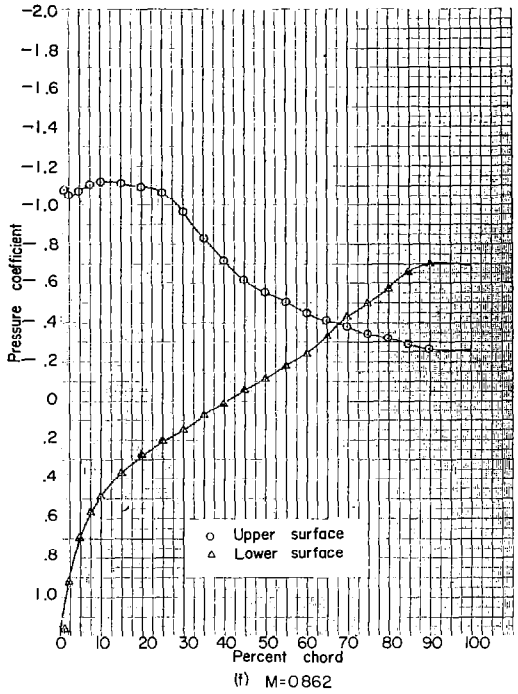


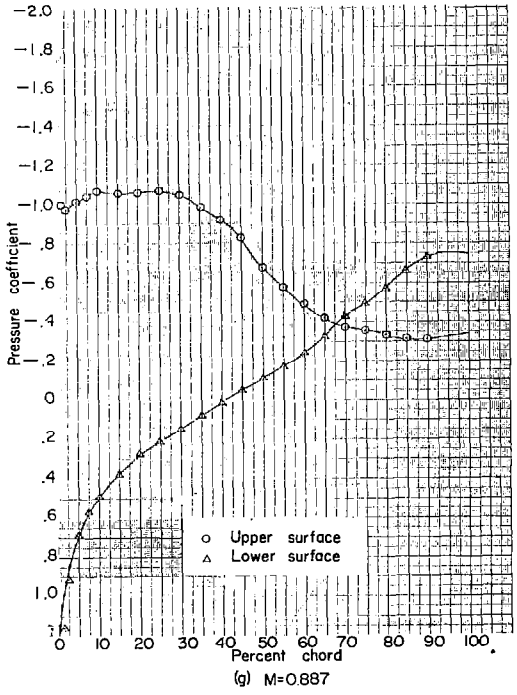
Figure 31.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 10^{\circ}$.



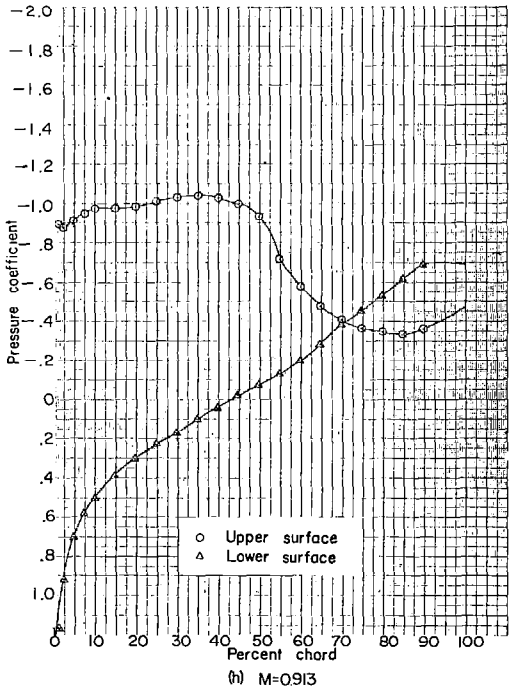
(e) M=0842



(f) M=0862

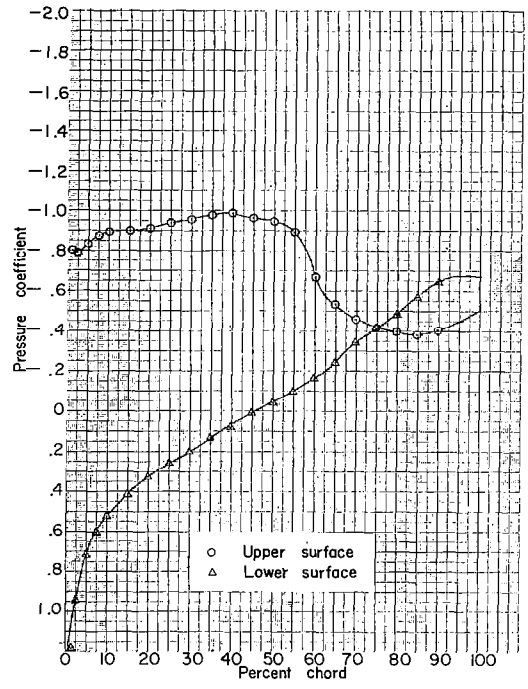


(g) M=0.887

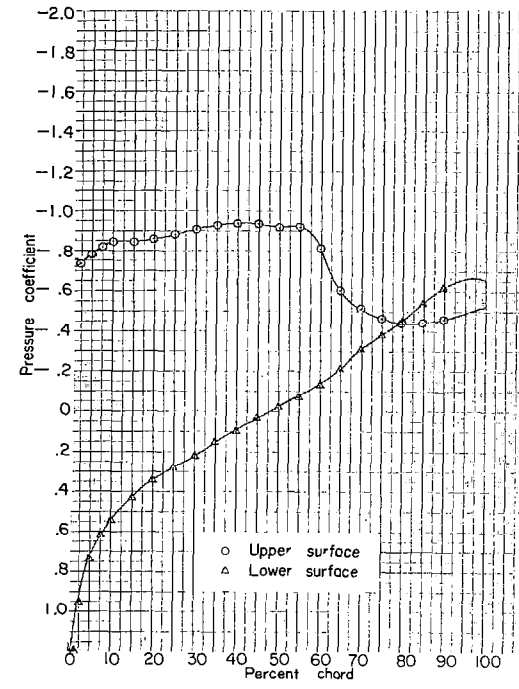


(h) M=0913

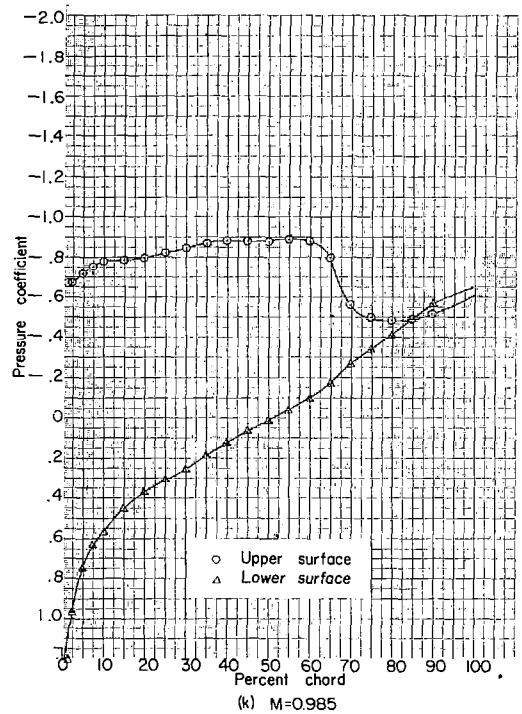
Figure 31.- Continued. NACA 16-012; $\alpha = 10^{\circ}$.



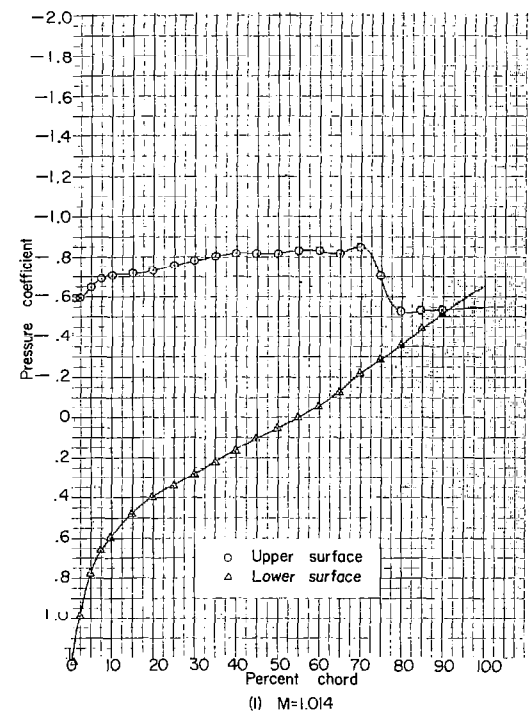
(i) M=0.936



(j) M=0.958

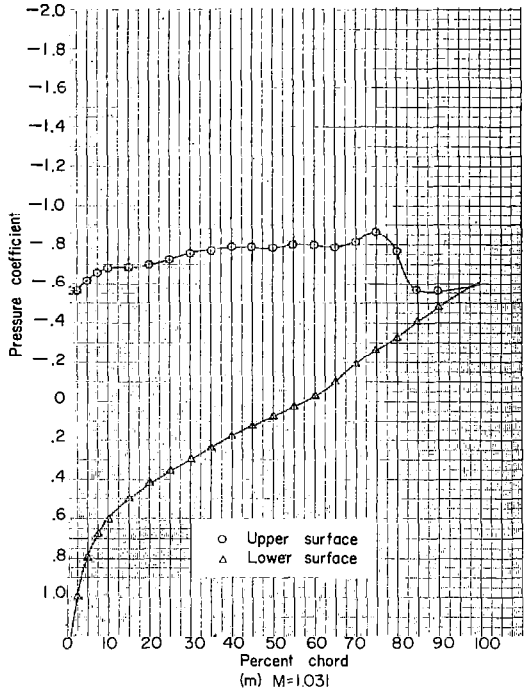


(k) M=0.985

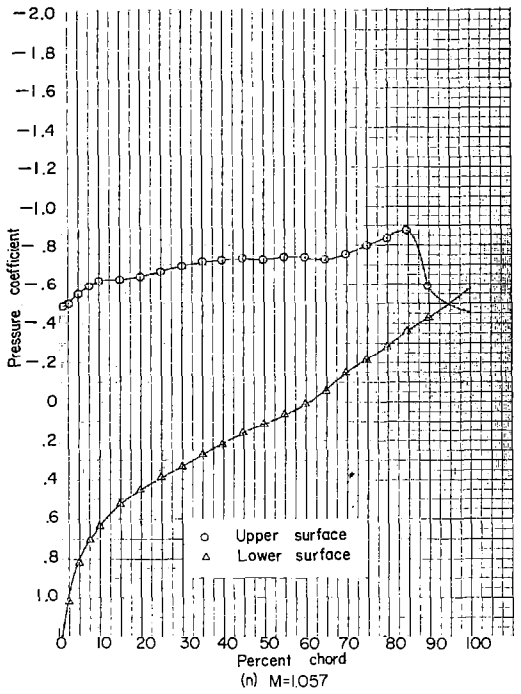


(l) M=1.014

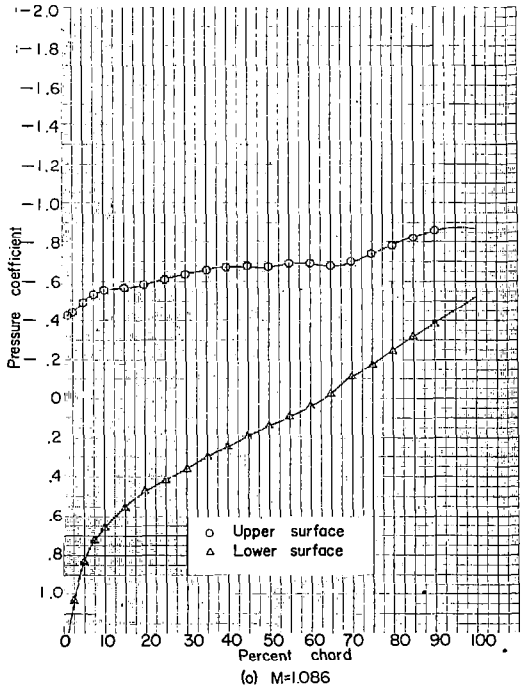
Figure 31.- Continued. NACA 16-012; $\alpha = 10^{\circ}$.



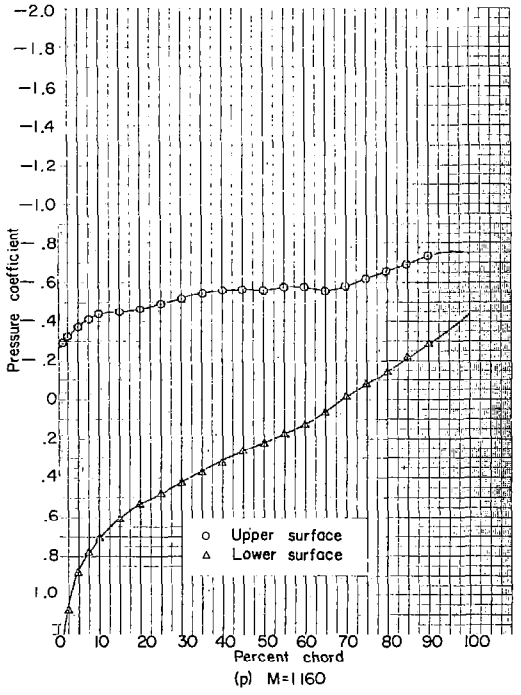
(m) M=1.031



(n) M=1.057



(o) M=1.086



(p) M=1.160

Figure 31.- Concluded. NACA 16-012; $\alpha = 10^{\circ}$.

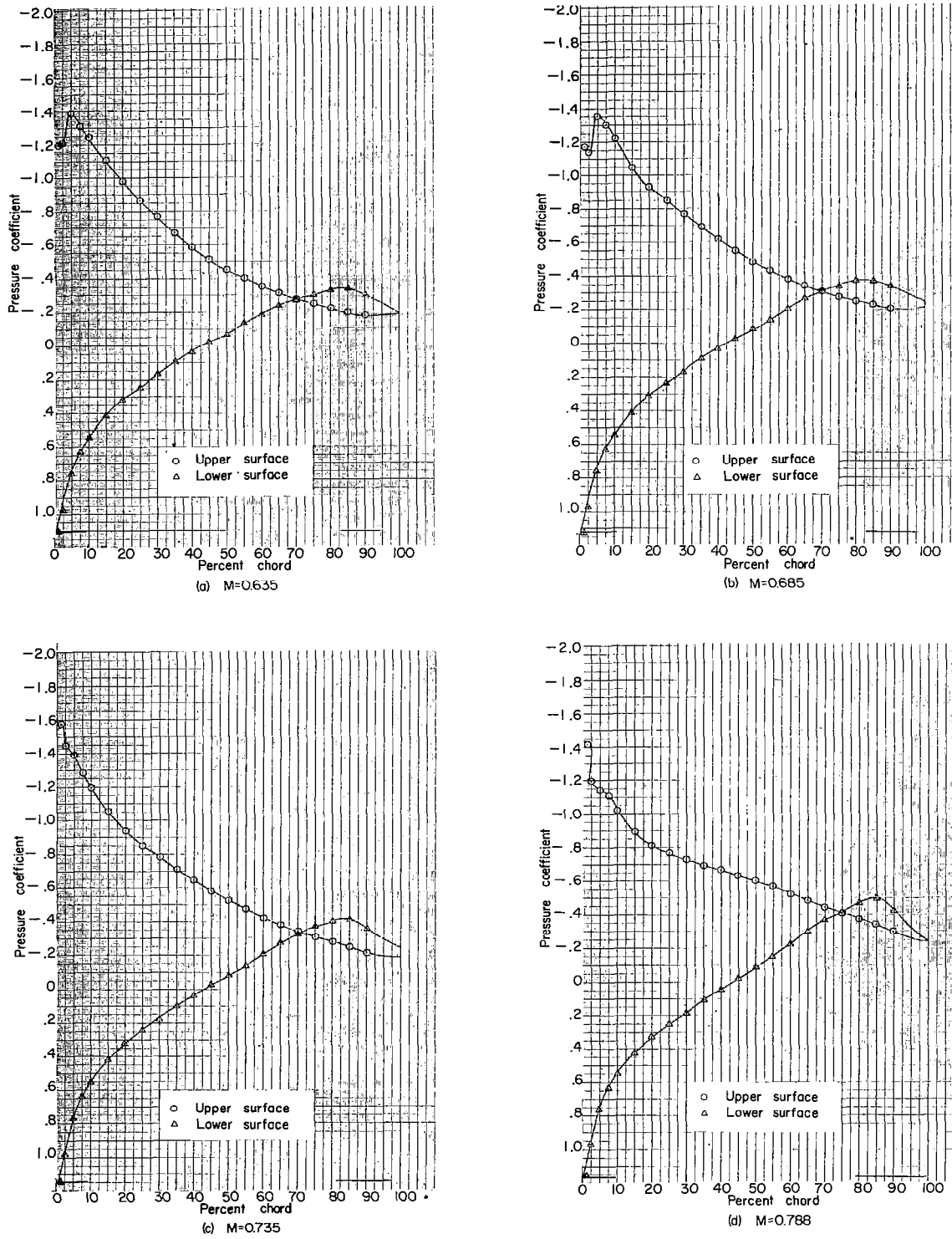
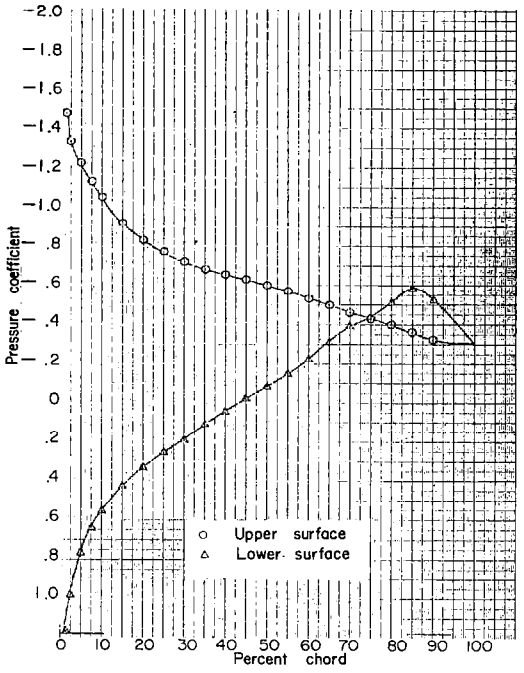
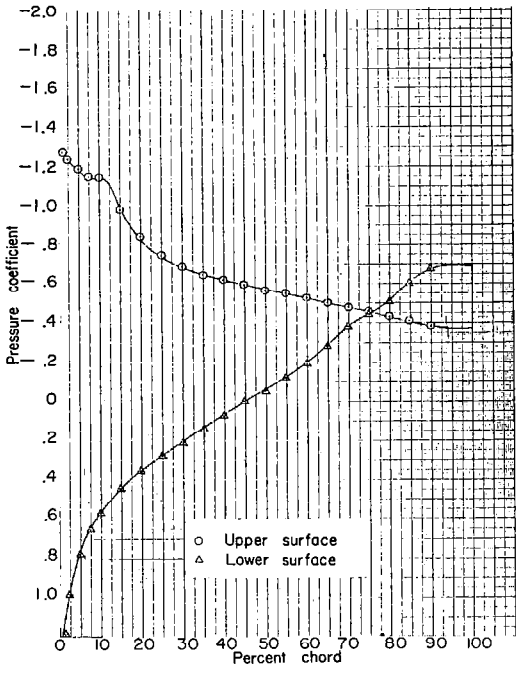


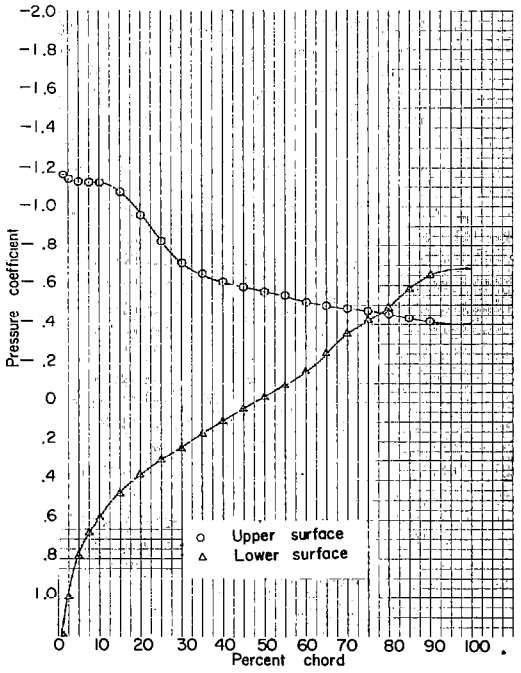
Figure 32.- Pressure distributions over NACA 16-012 airfoil section. $\alpha = 12^{\circ}$.



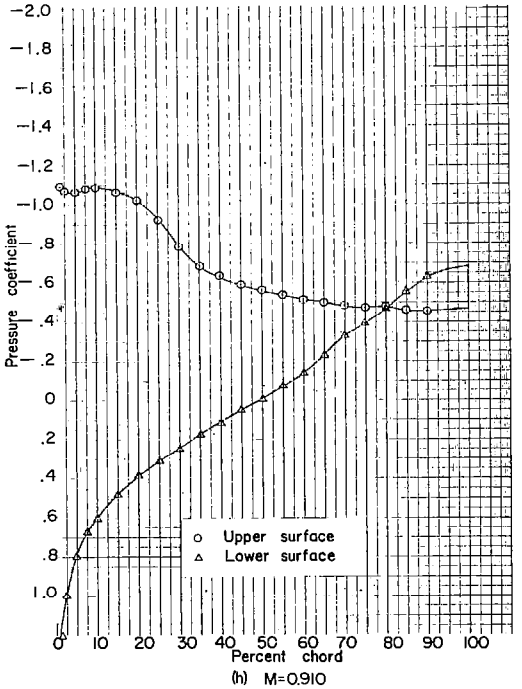
(e) M=0.815



(f) M=0.861

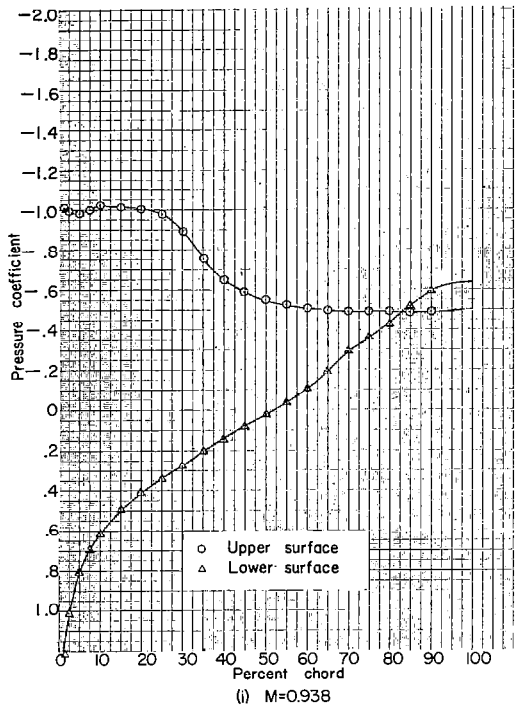


(g) M=0.890

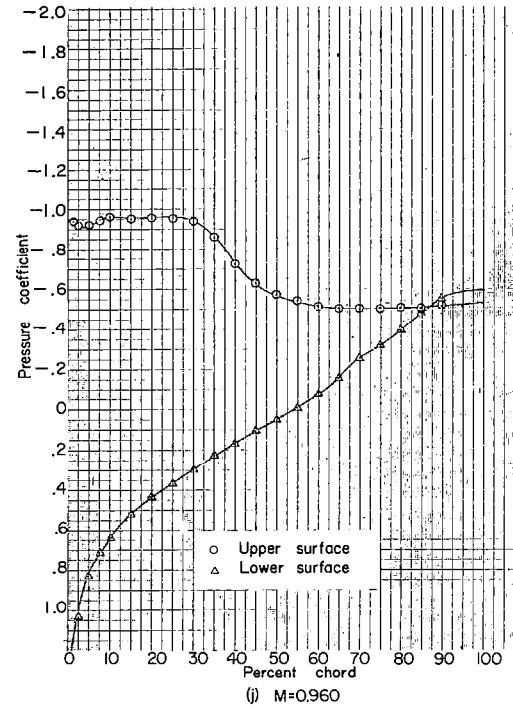


(h) M=0.910

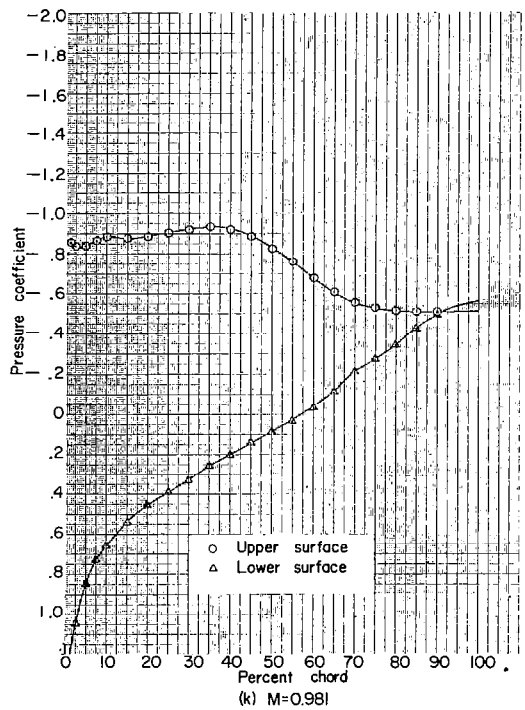
Figure 32.- Continued. NACA 16-012; $\alpha = 12^{\circ}$.



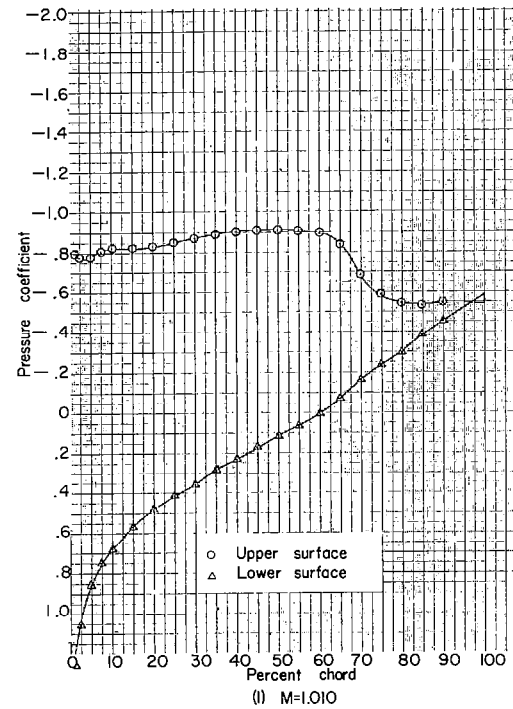
(i) M=0.938



(j) M=0.960

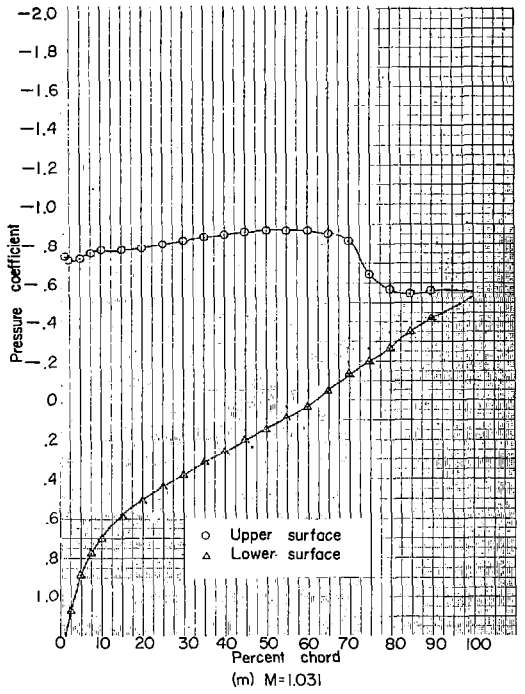


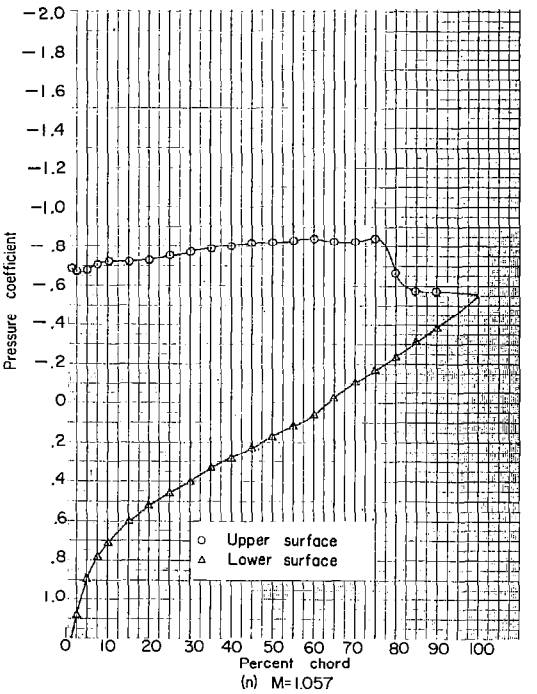
(k) M=0.981

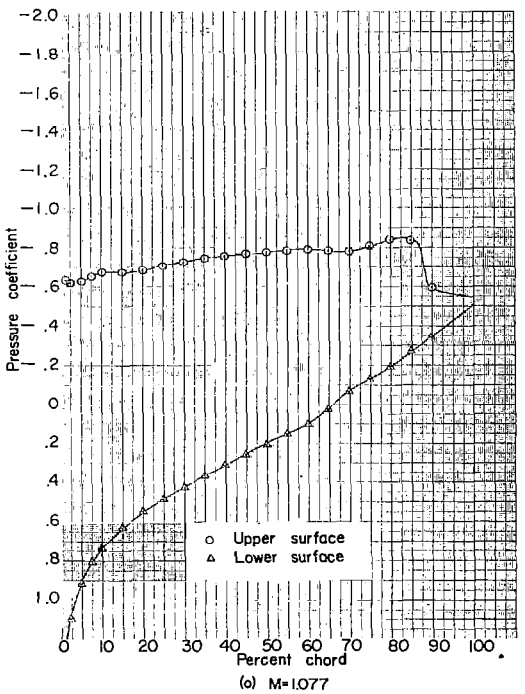


(l) M=1.010

Figure 32.- Continued. NACA 16-012; $\alpha = 12^{\circ}$.







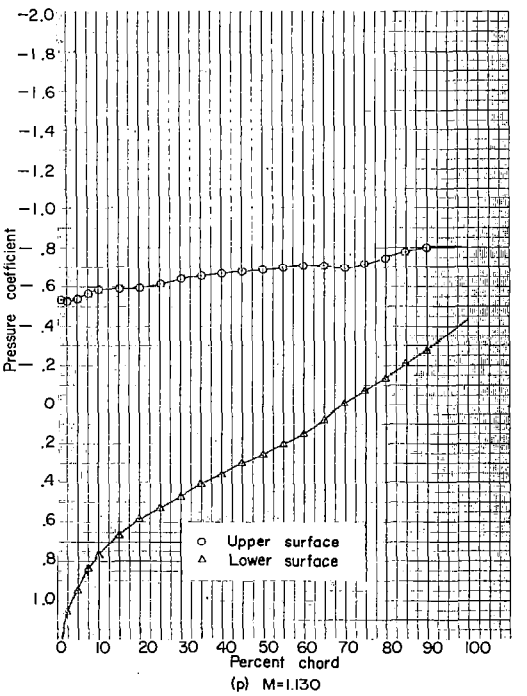


Figure 32.- Concluded. NACA 16-012; $\alpha = 12^{\circ}$.

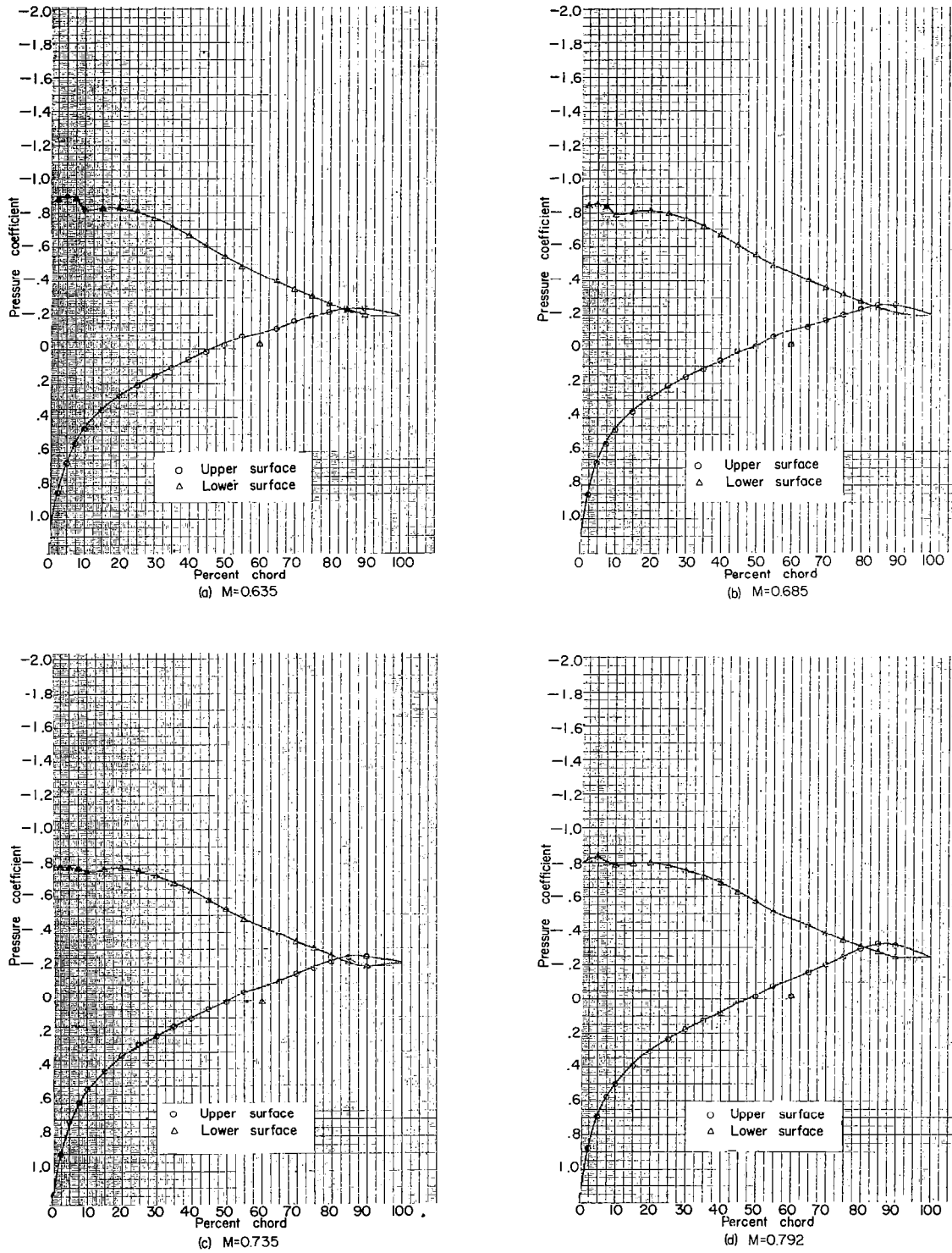
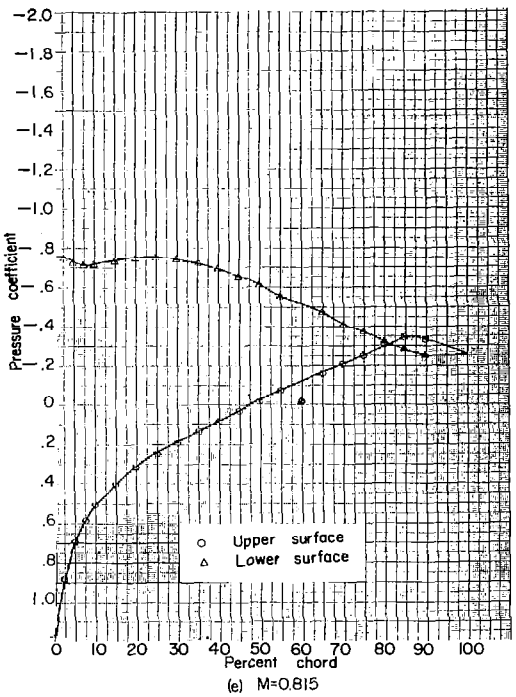
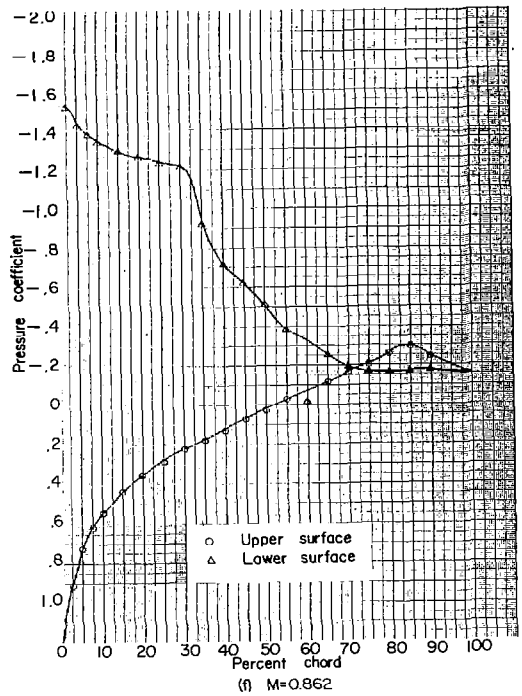


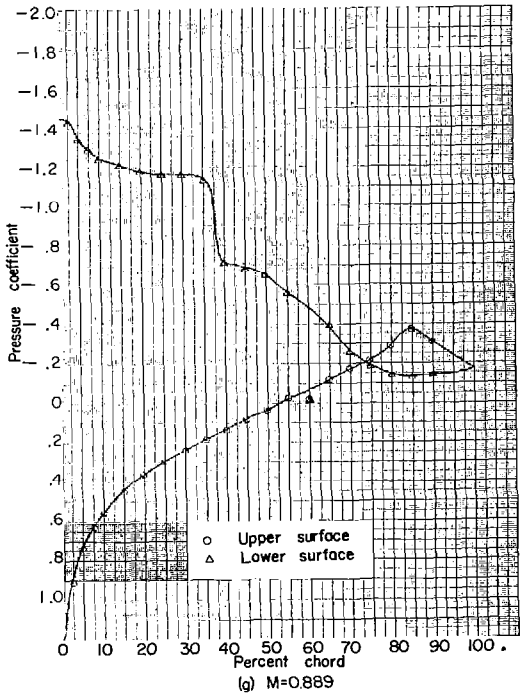
Figure 33.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = -10^{\circ}$.



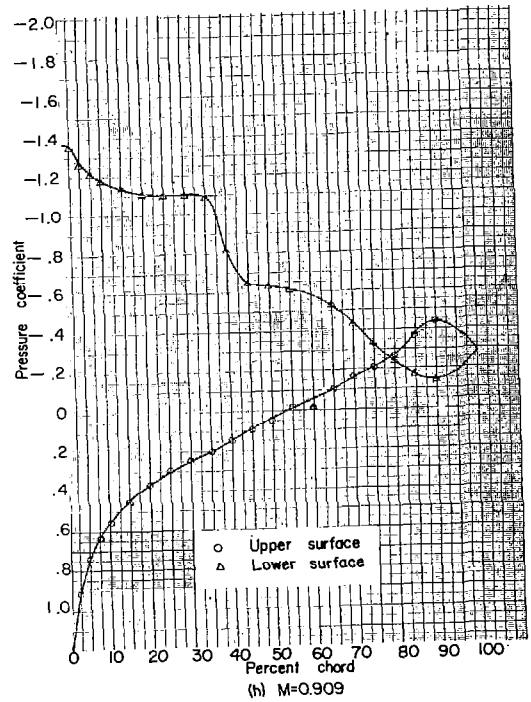
(e) M=0.815



(f) M=0.862

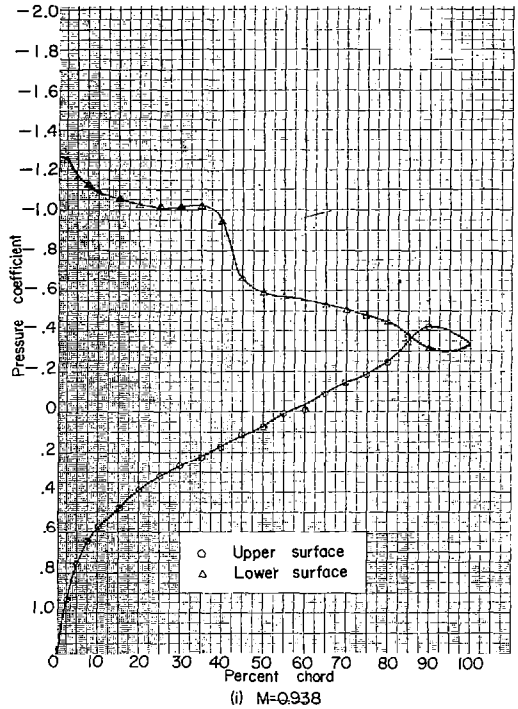


(g) M=0.889

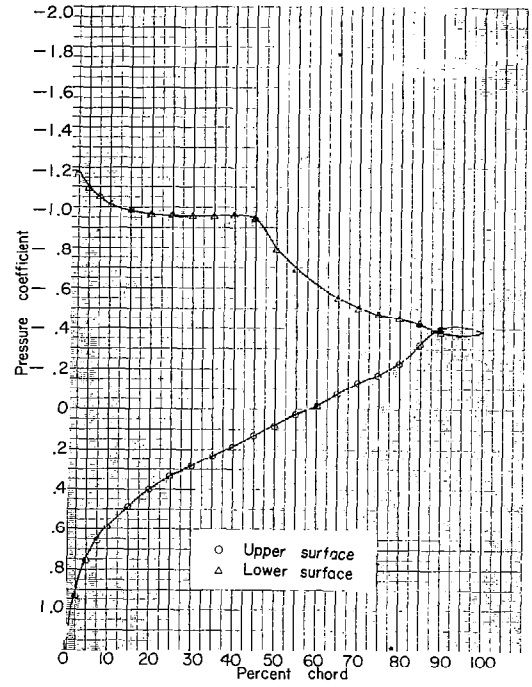


(h) M=0.909

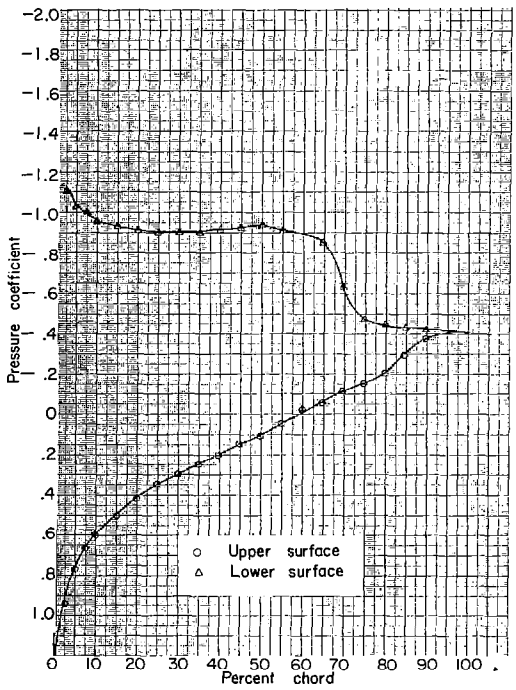
Figure 33.- Continued. NACA 16-206; $\alpha = -10^{\circ}$.



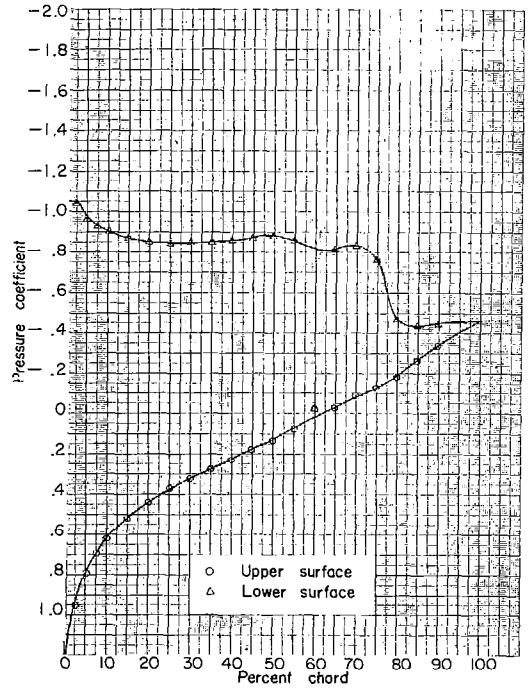
(i) M=0.938



(j) M=0.963

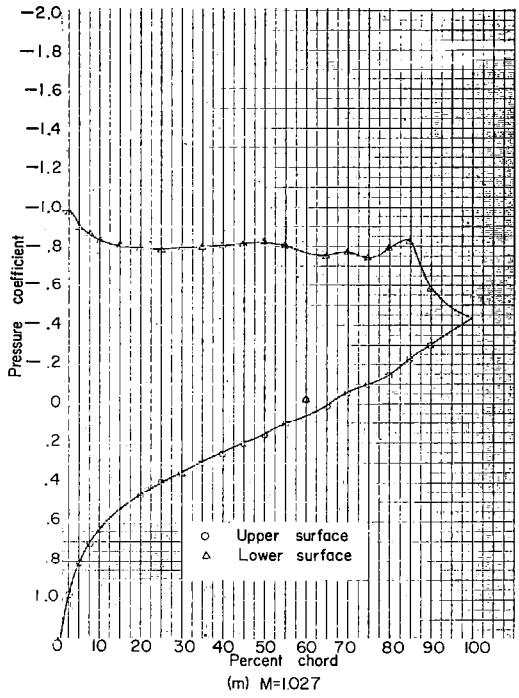


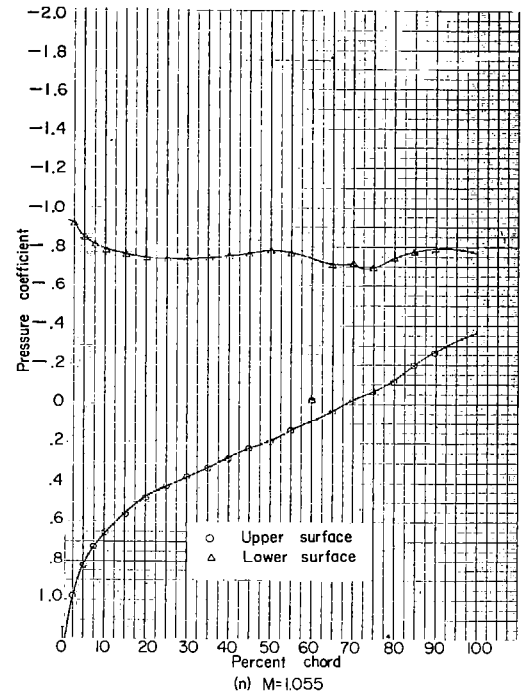
(k) M=0.983

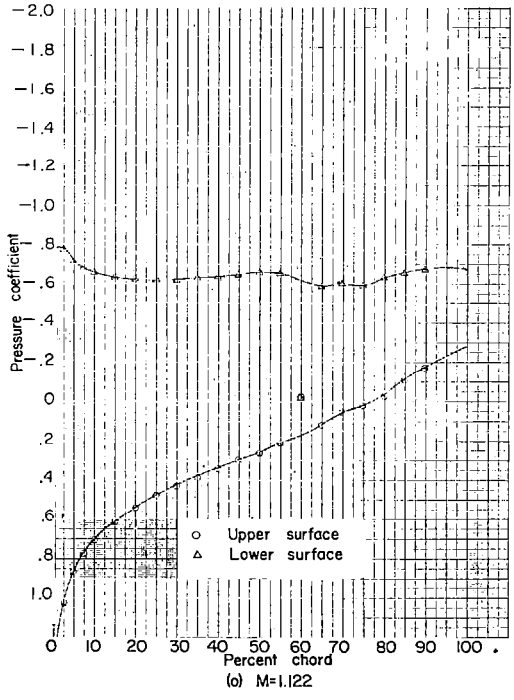


(l) M=1.006

Figure 33.- Continued. NACA 16-206; $\alpha = -10^{\circ}$.







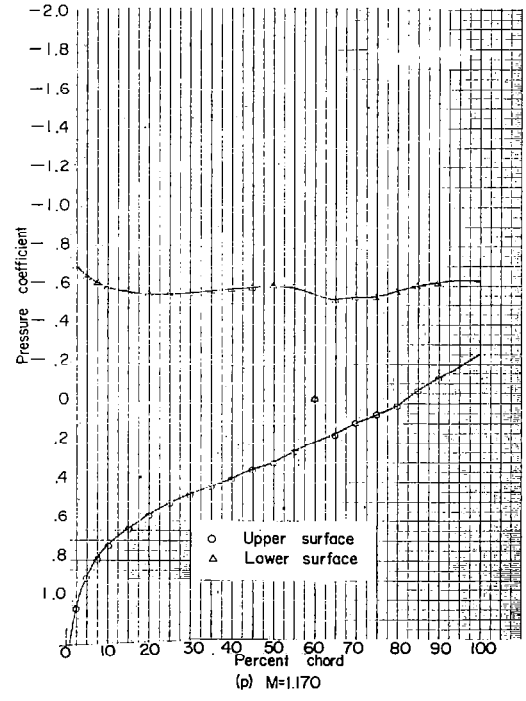


Figure 33.- Concluded. NACA 16-206; $\alpha = -10^{\circ}$.

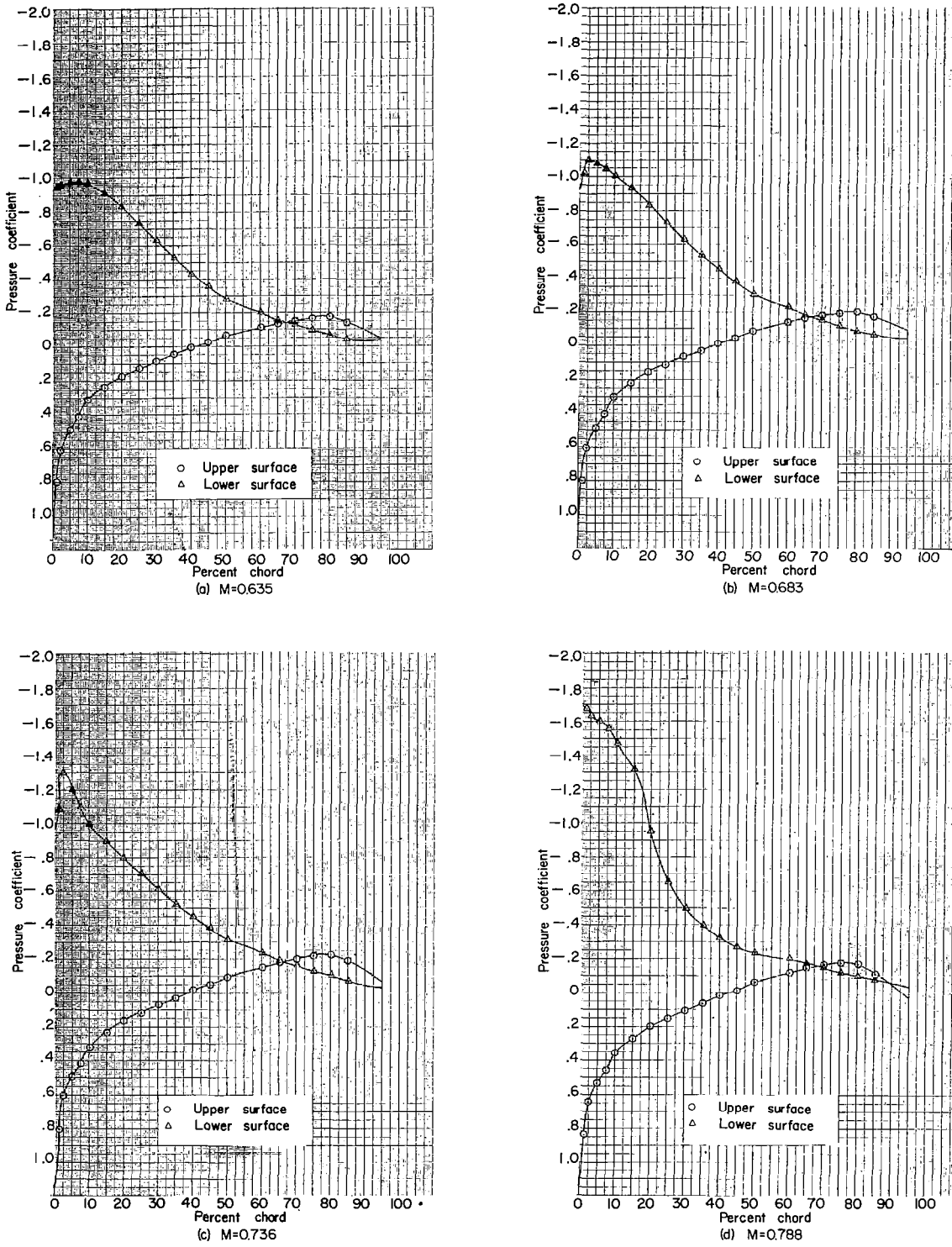
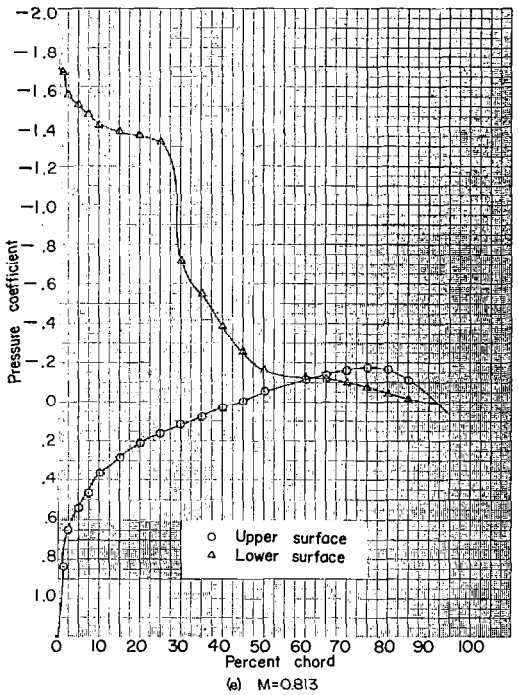
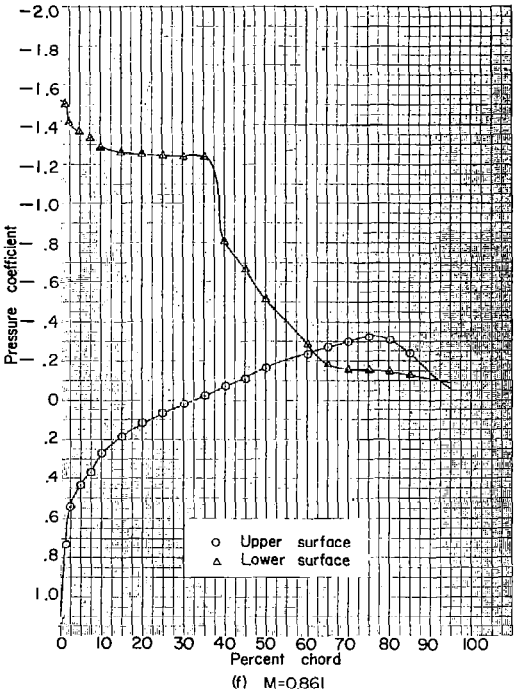


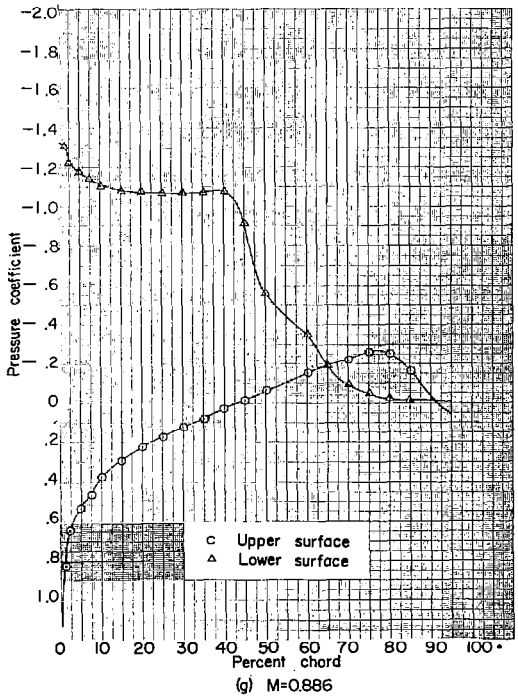
Figure 34.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = -8^{\circ}$.



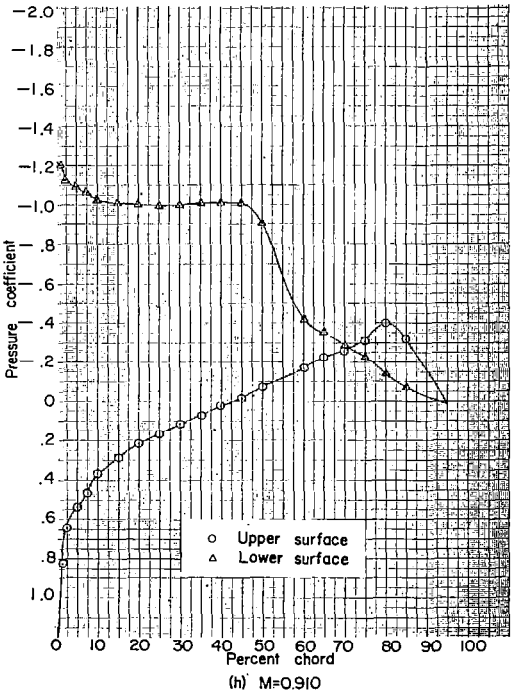
(e) M=0.813



(f) M=0.861

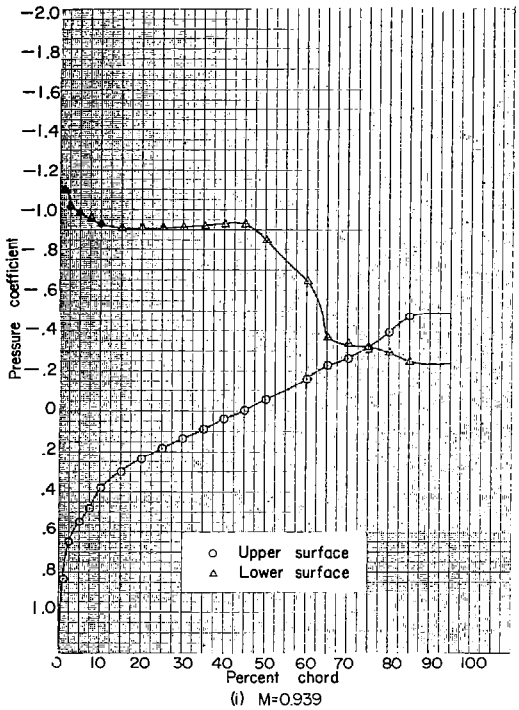


(g) M=0.886

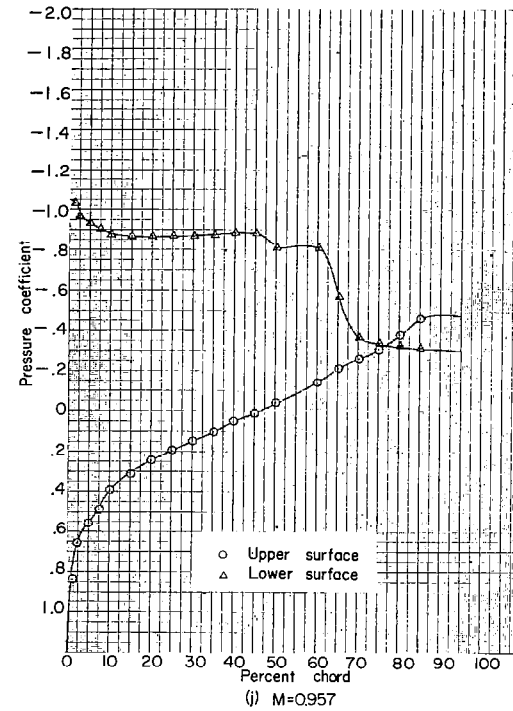


(h) M=0.910

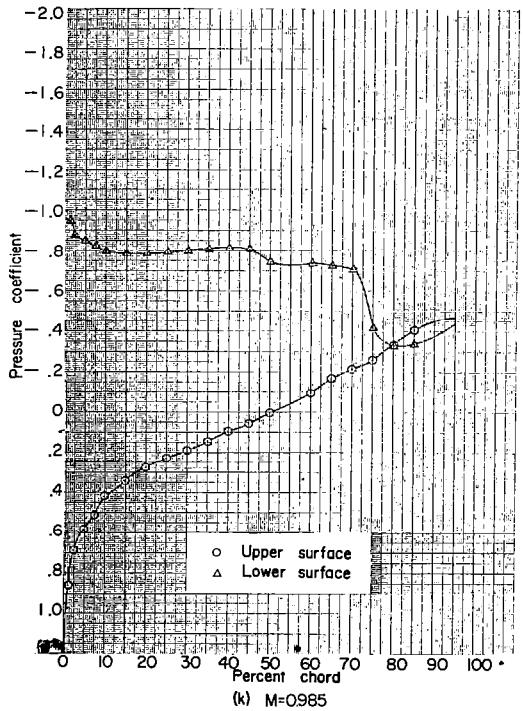
Figure 34.- Continued. NACA 16-206; $\alpha = -8^{\circ}$.



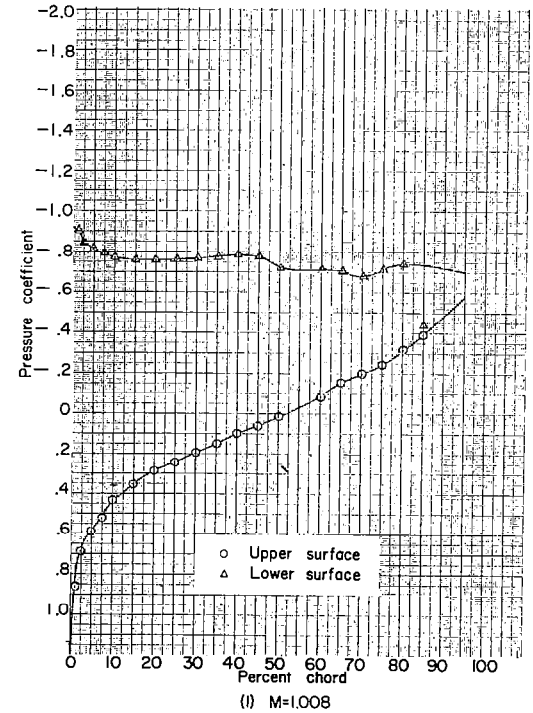
(i) M=0.939



(j) M=0.957

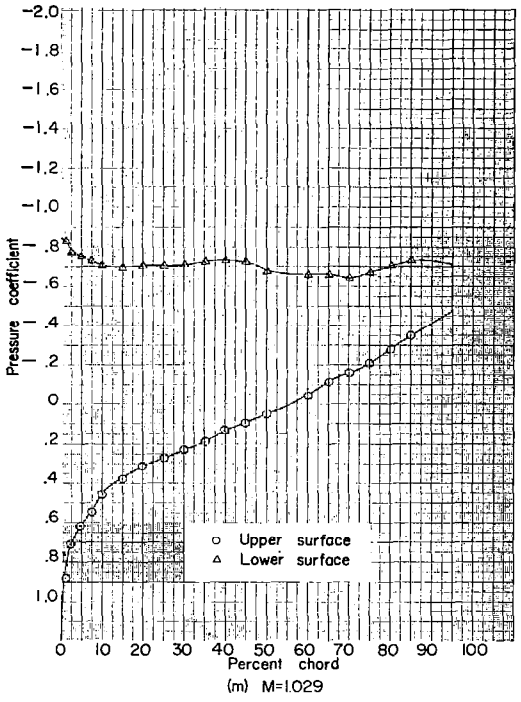


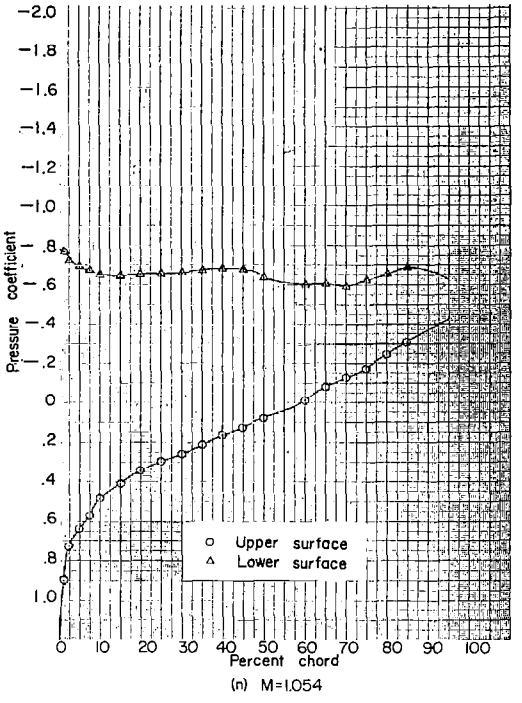
(k) M=0.985

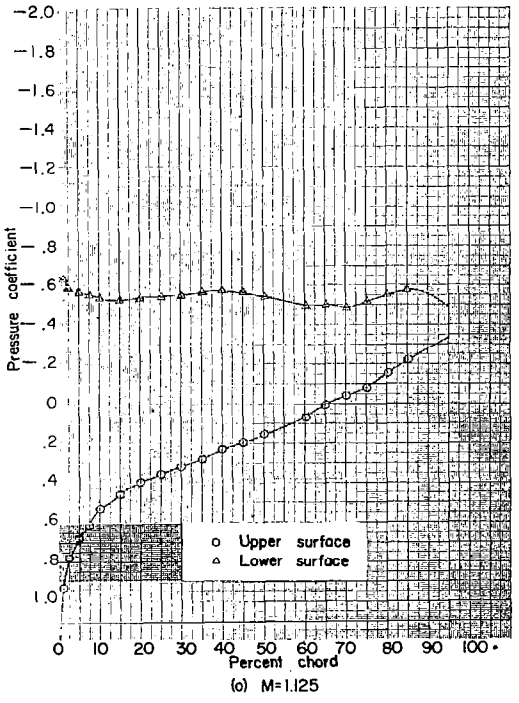


(l) M=1.008

Figure 34.- Continued. NACA 16-206; $\alpha = -8^{\circ}$.







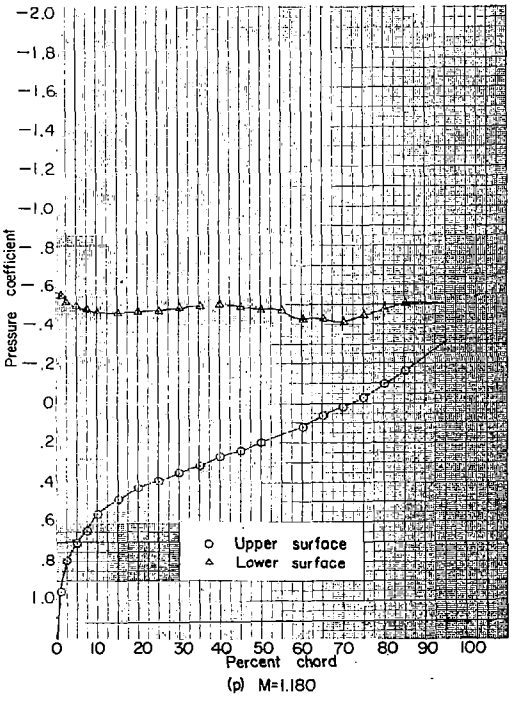


Figure 34.- Concluded. NACA 16-206; $\alpha = -8^{\circ}$.

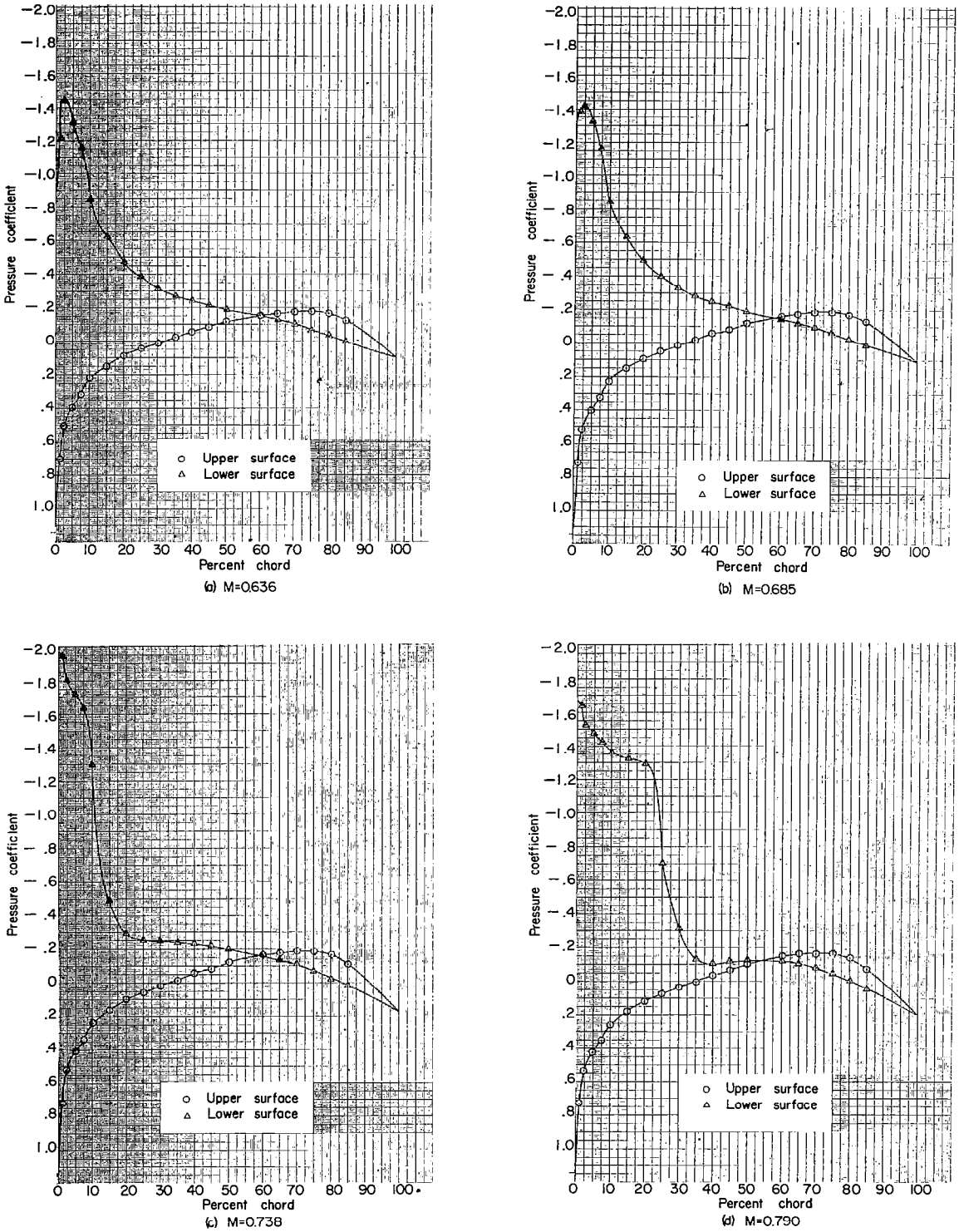
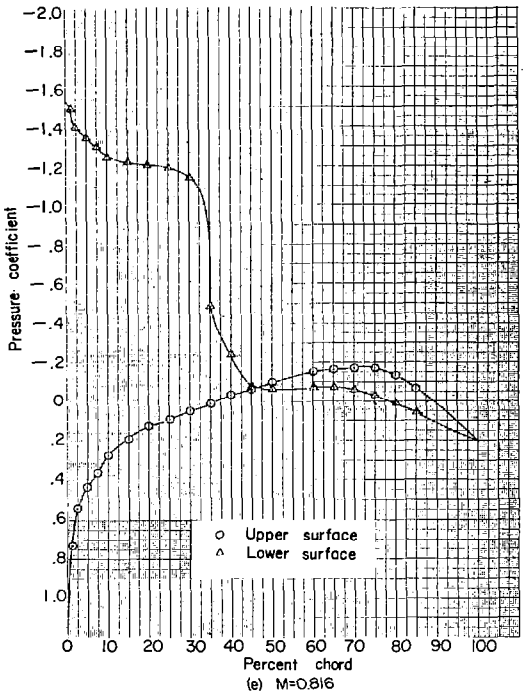
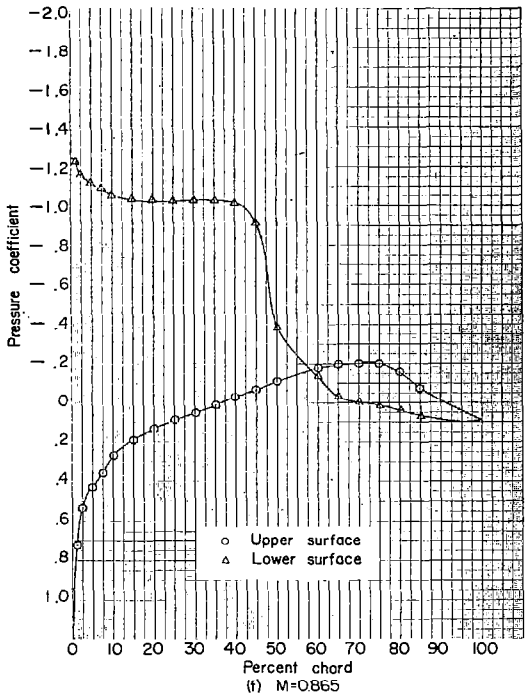


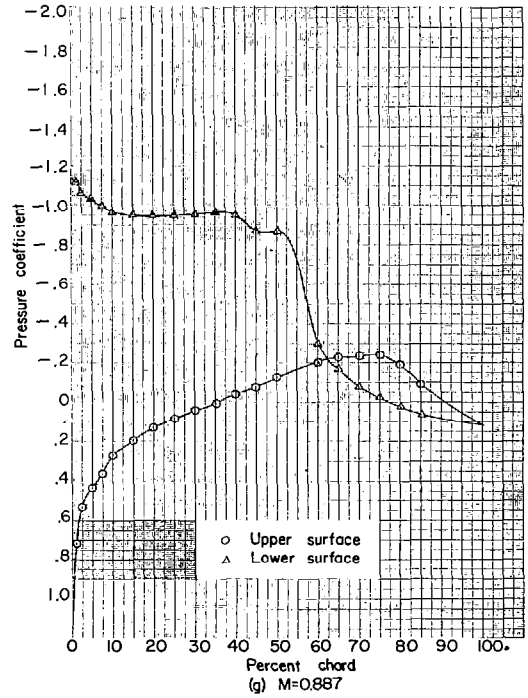
Figure 35.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = -6^{\circ}$.



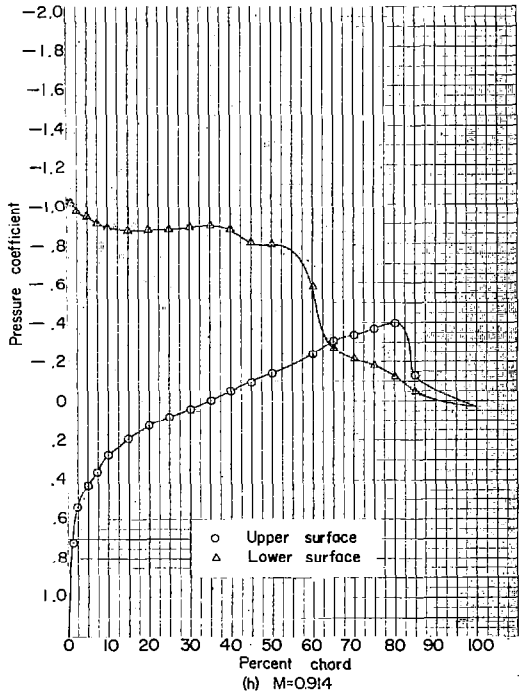
(e) M=0.816



(f) M=0.865

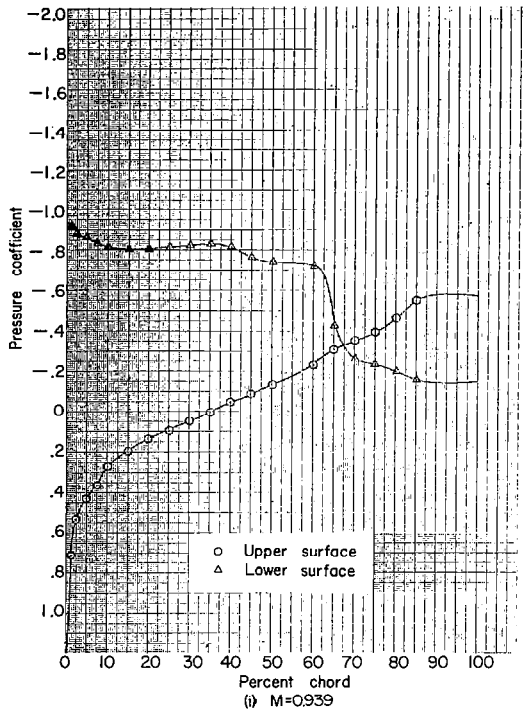


(g) M=0.887

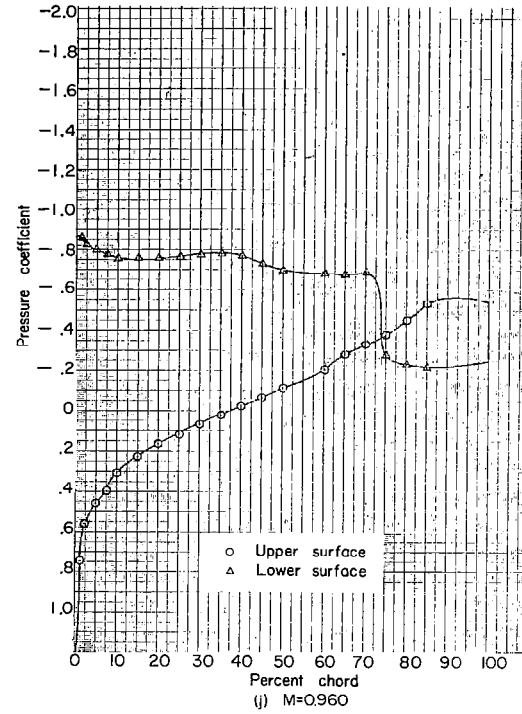


(h) M=0.914

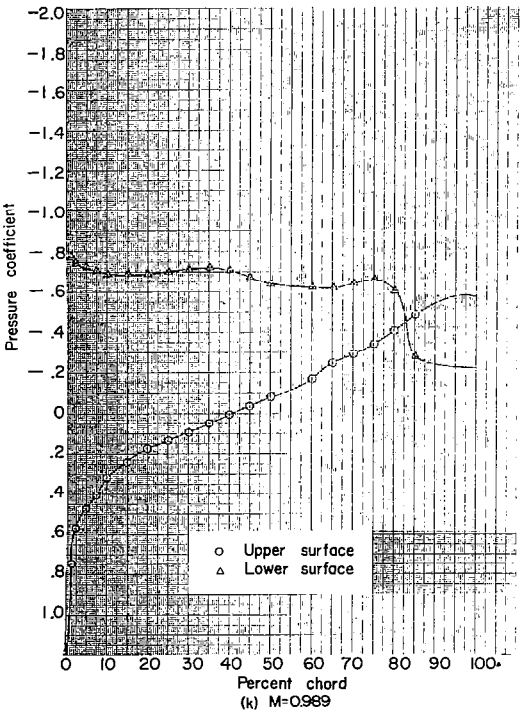
Figure 35.- Continued. NACA 16-206; $\alpha = -6^{\circ}$.



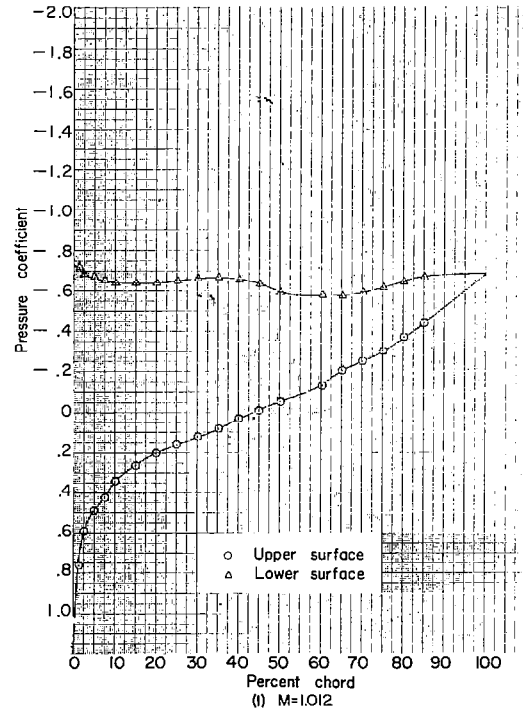
(i) M=0.939



(j) M=0.960

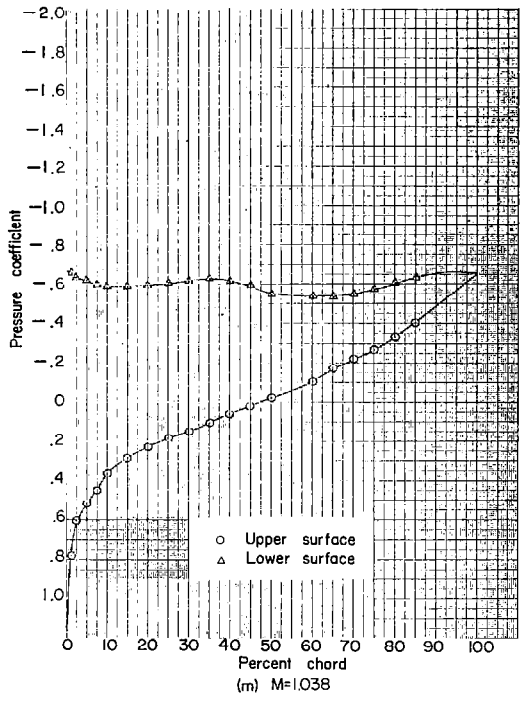


(k) M=0.989

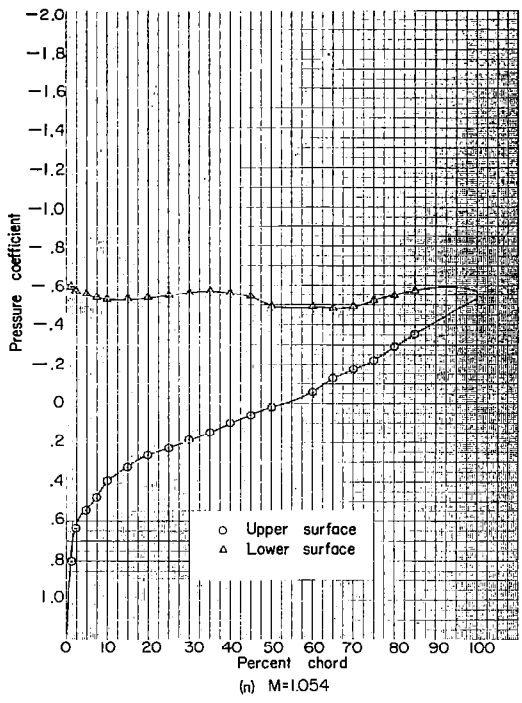


(l) M=1.012

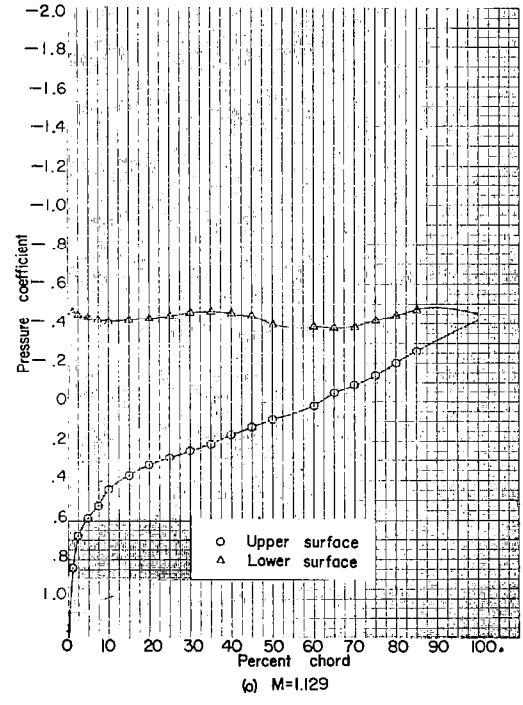
Figure 35.- Continued. NACA 16-206; $\alpha = -6^{\circ}$.



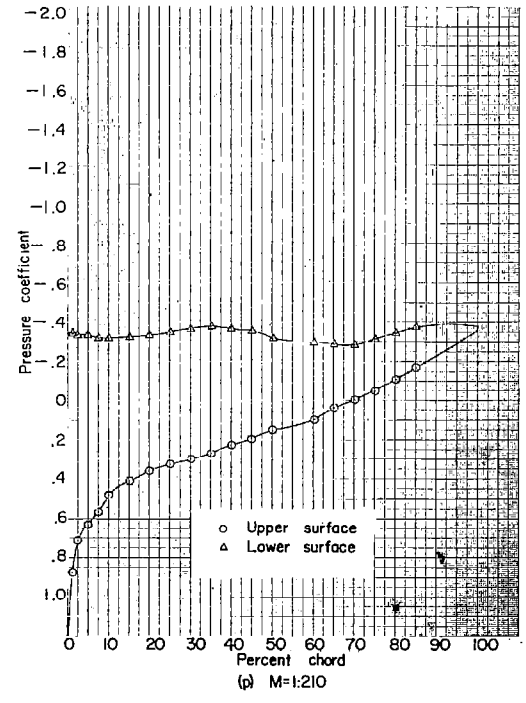
(m) M=1.038



(n) M=1.054



(o) M=1.129



(p) M=1.210

Figure 35.- Concluded. NACA 16-206; $\alpha = -6^{\circ}$.

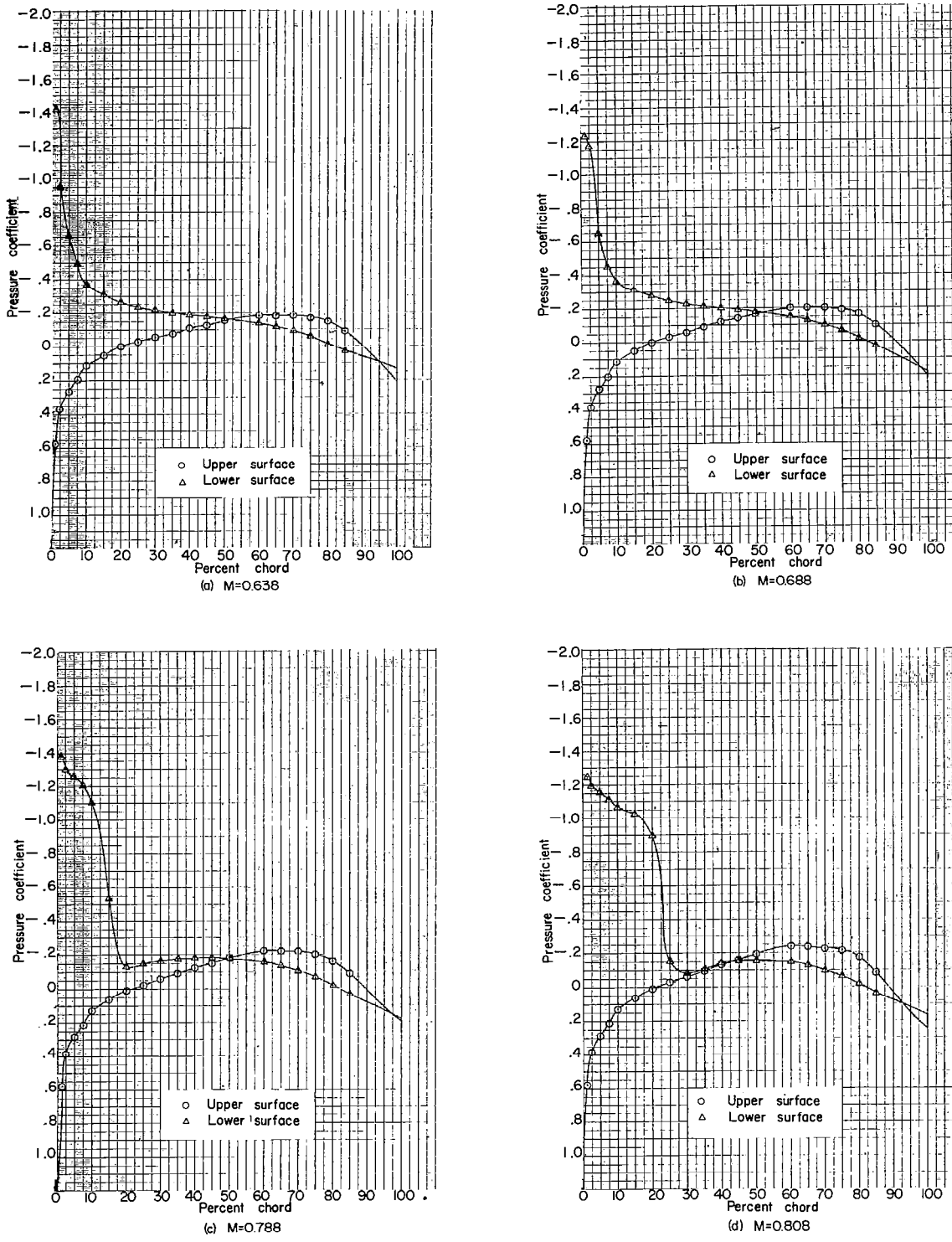
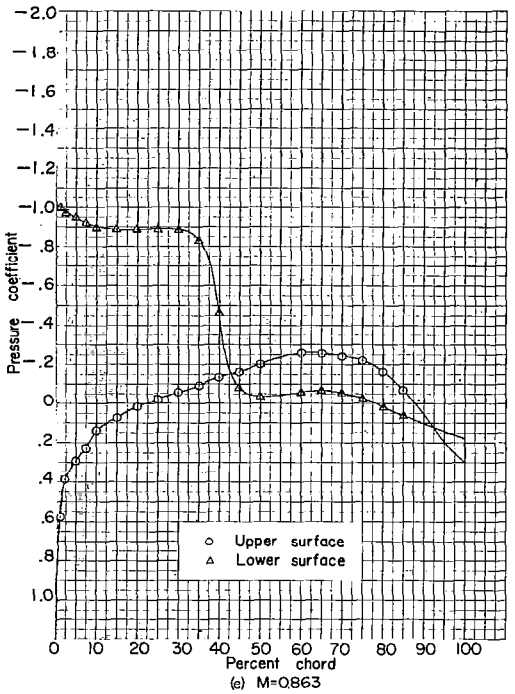
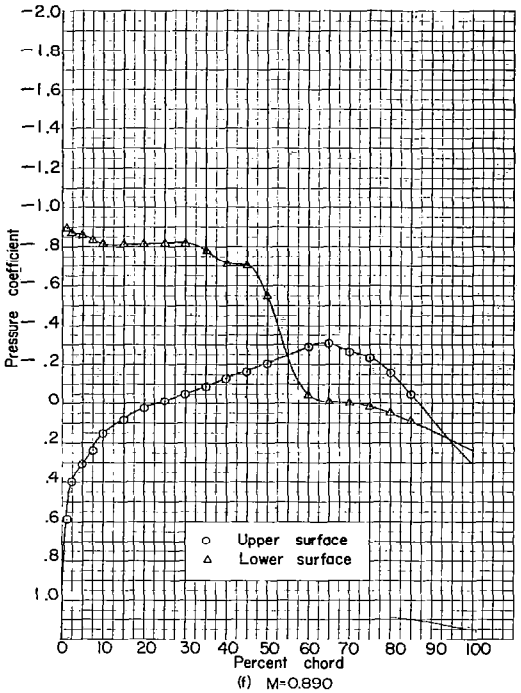
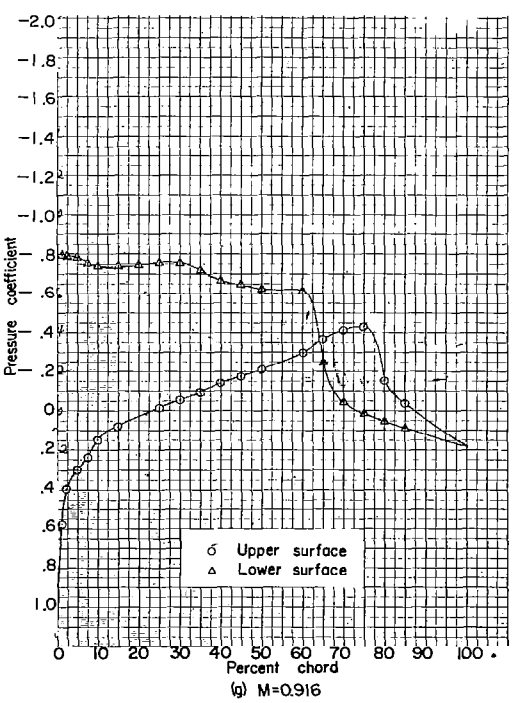


Figure 36.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = -4^{\circ}$.







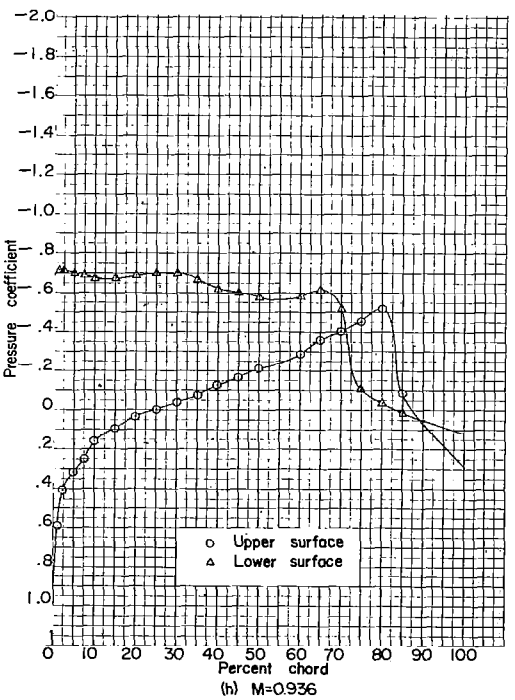
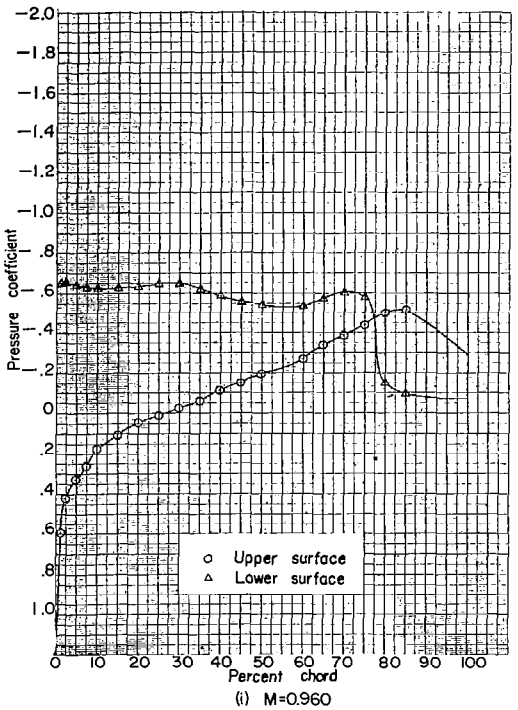
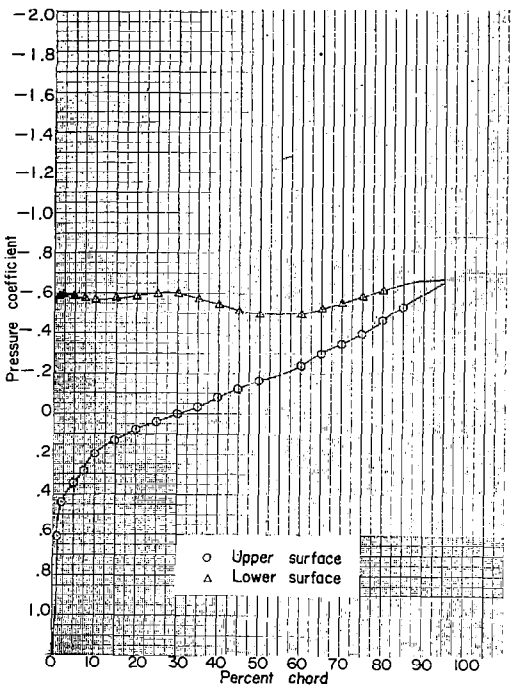


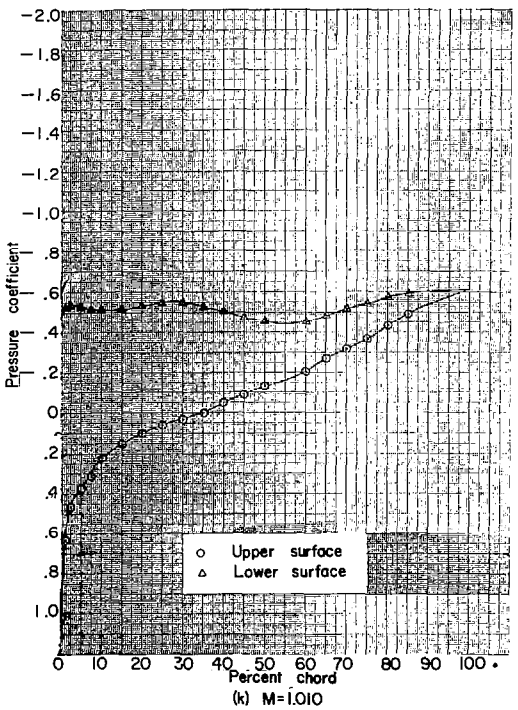
Figure 36.- Continued. NACA 16-206; $\alpha = -4^{\circ}$.



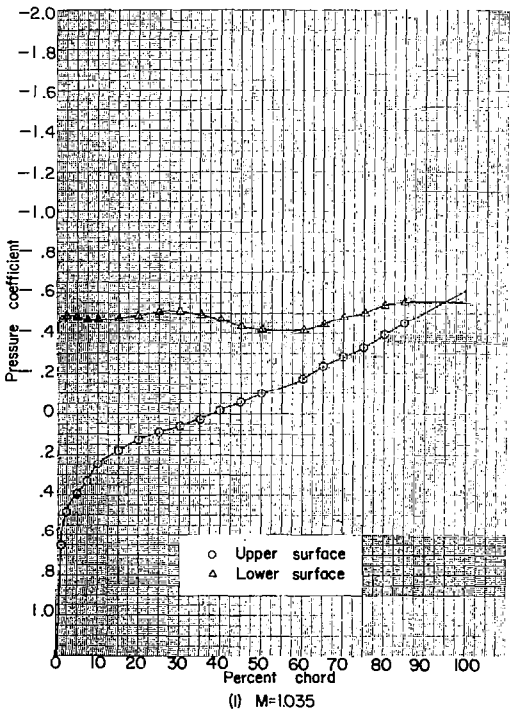
(i) M=0.960



(j) M=0.985

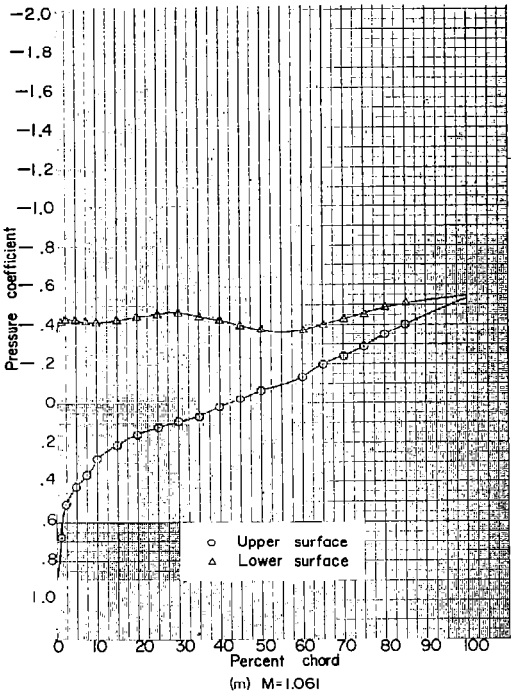


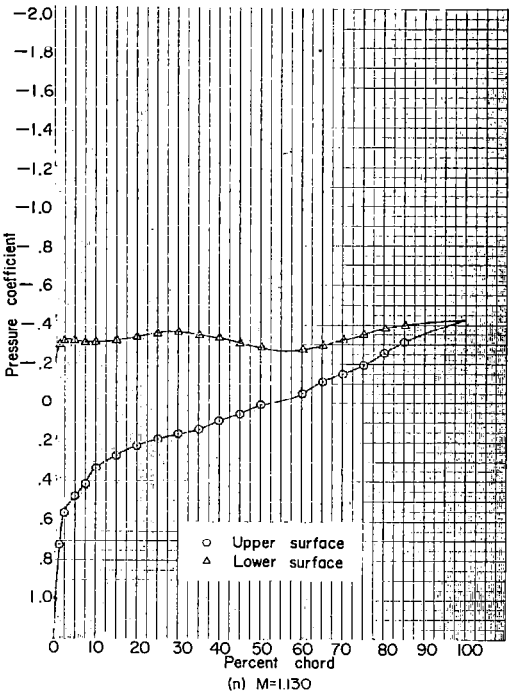
(k) M=1.010

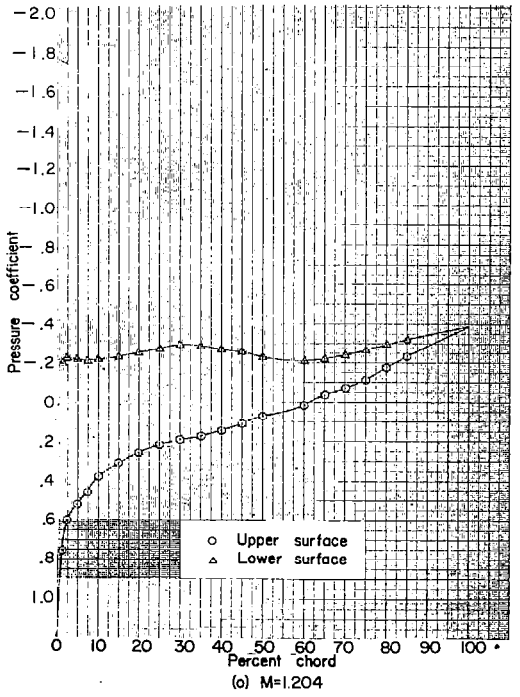


(l) M=1.035

Figure 36.- Continued. NACA 16-206; $\alpha = -4^{\circ}$.







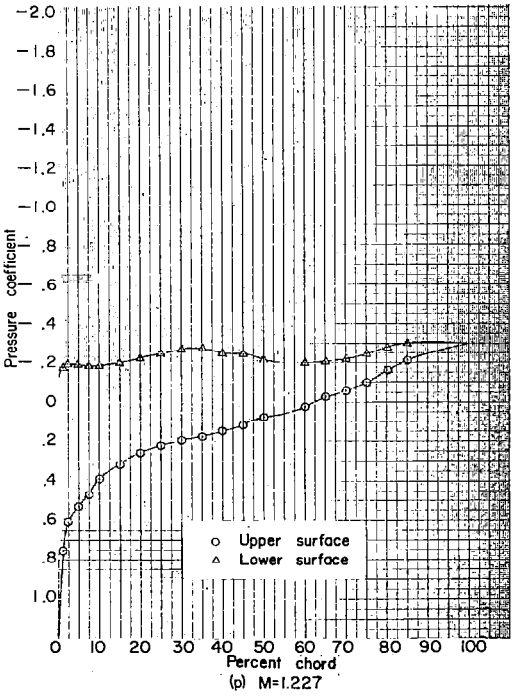


Figure 36.- Concluded. NACA 16-206; $\alpha = -4^{\circ}$.

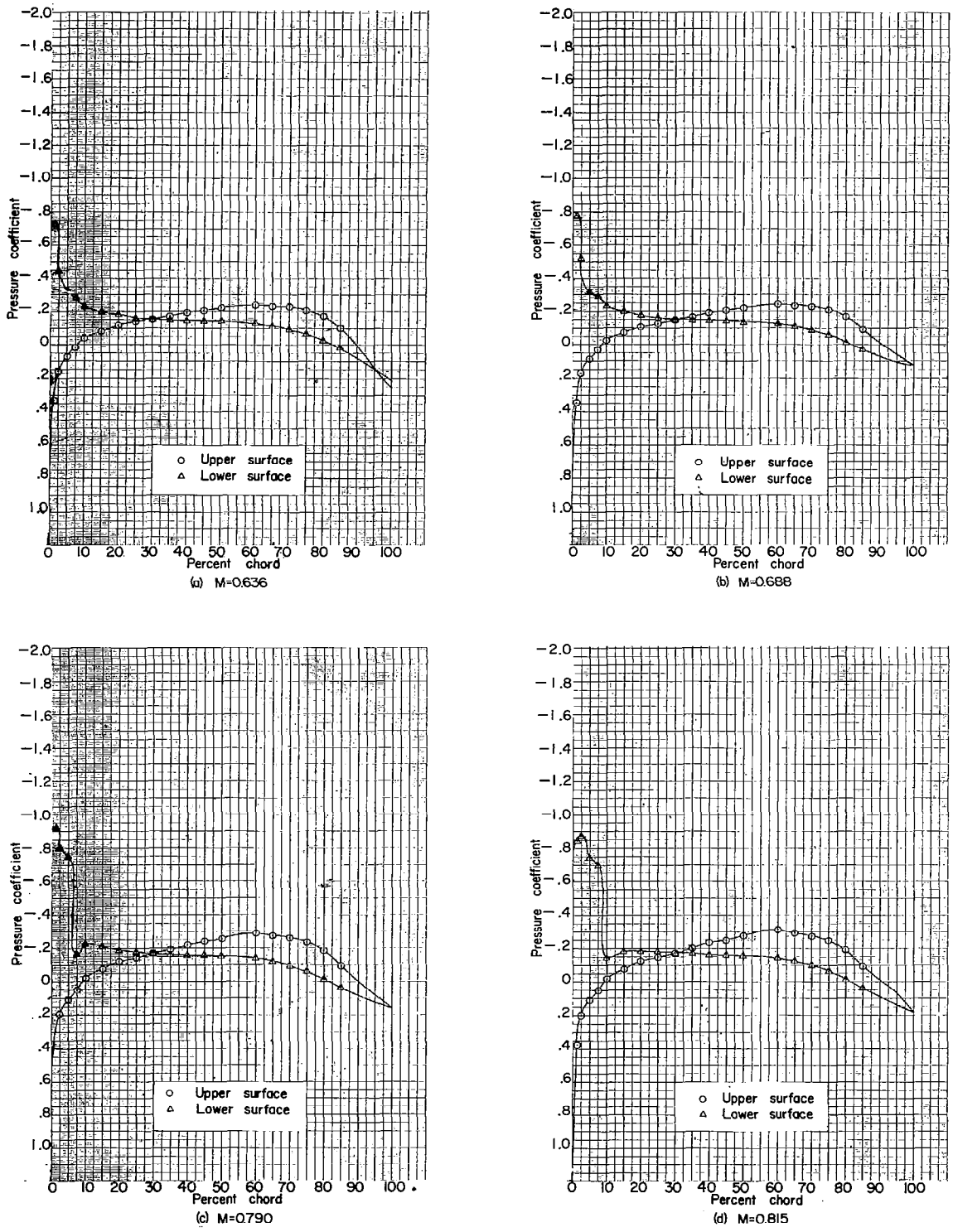
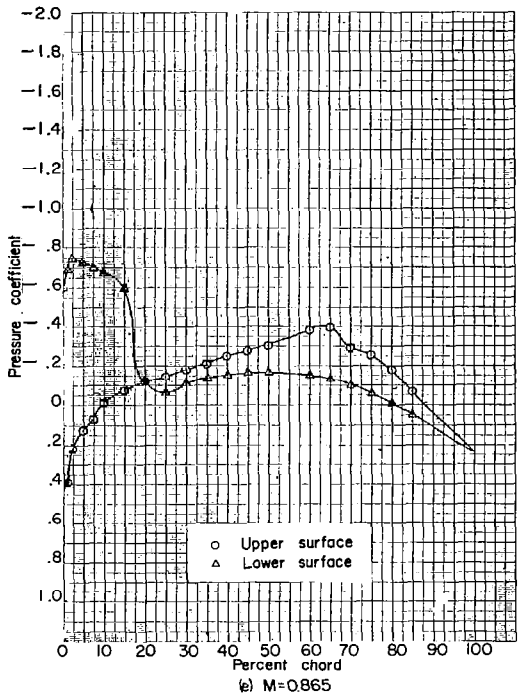
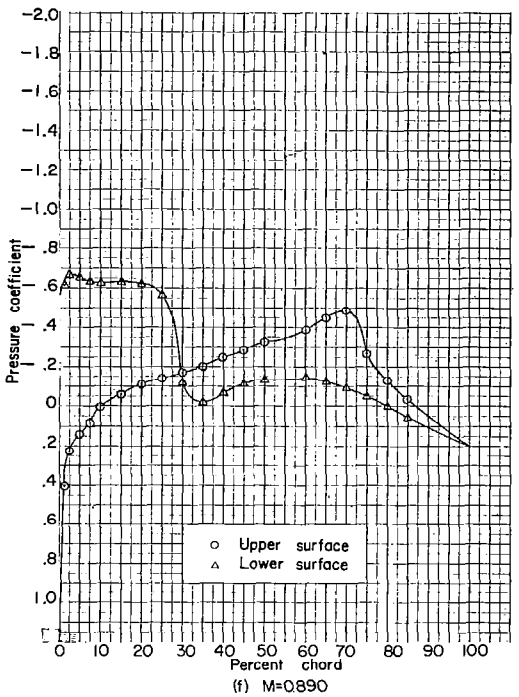


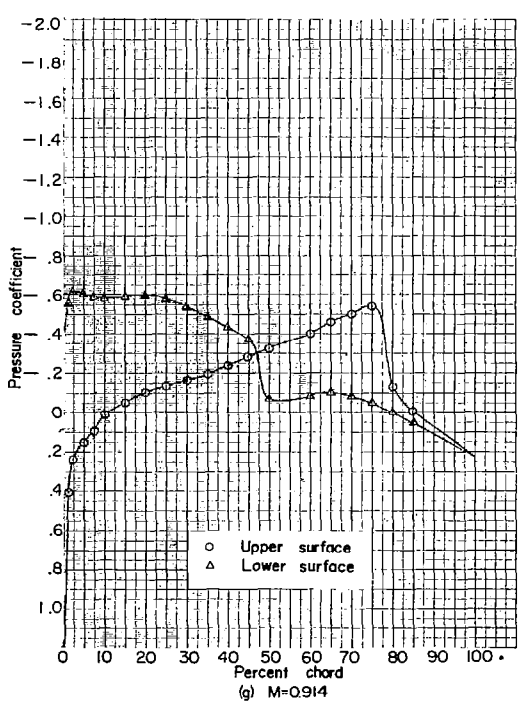
Figure 37.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = -2^{\circ}$.



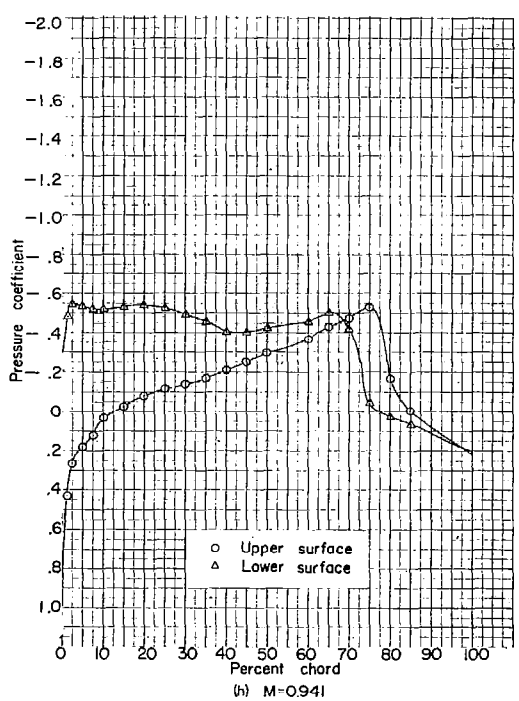
(e) M=0.865



(f) M=0.890

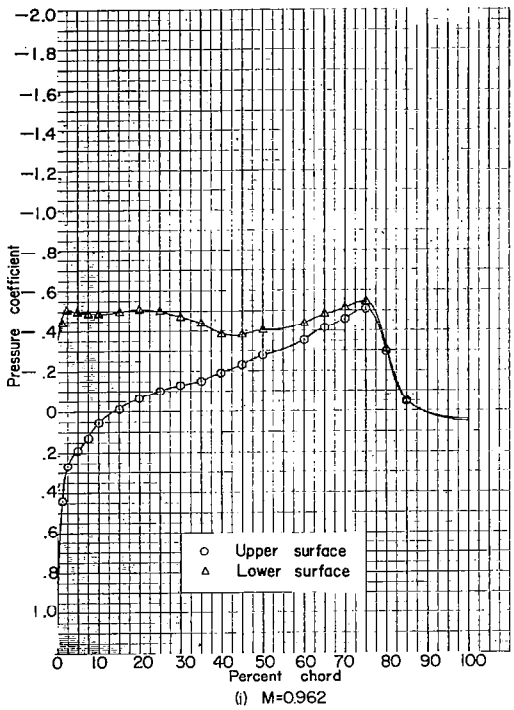


(g) M=0.914

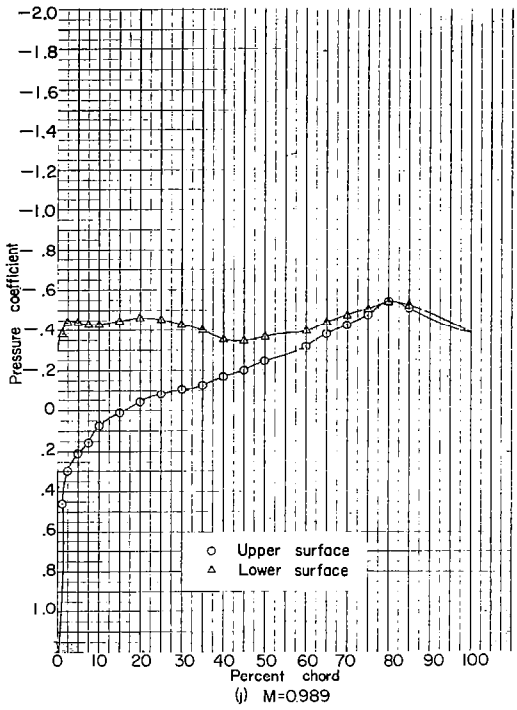


(h) M=0.941

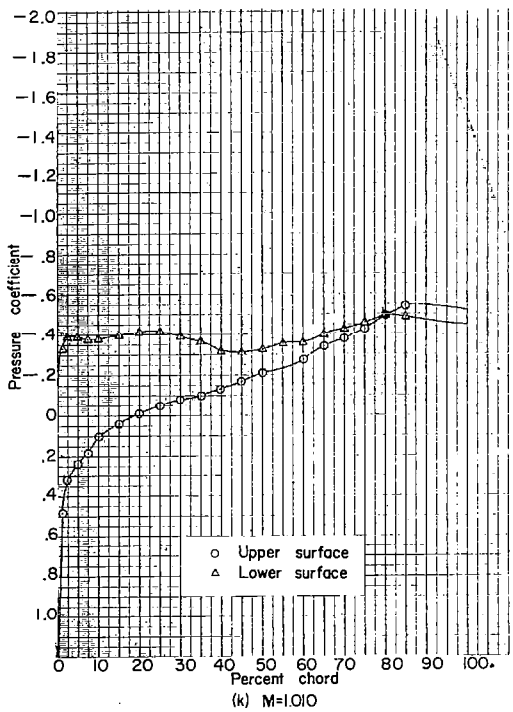
Figure 37.- Continued. NACA 16-206; $\alpha = -2^{\circ}$.



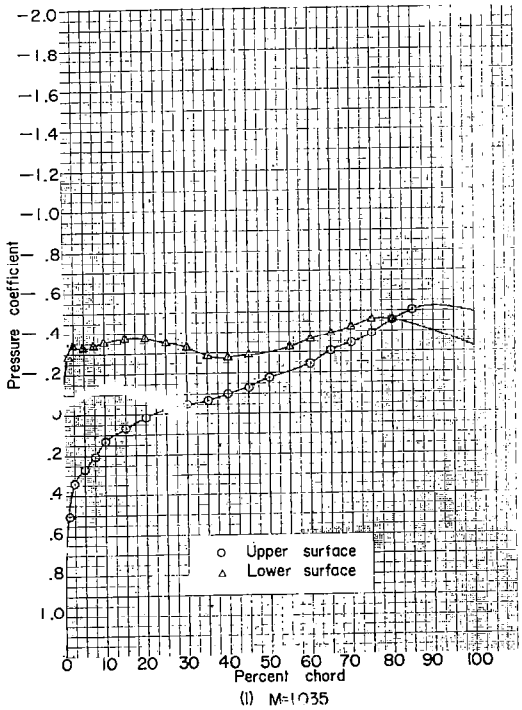
(i) M=0.962



(j) M=0.989

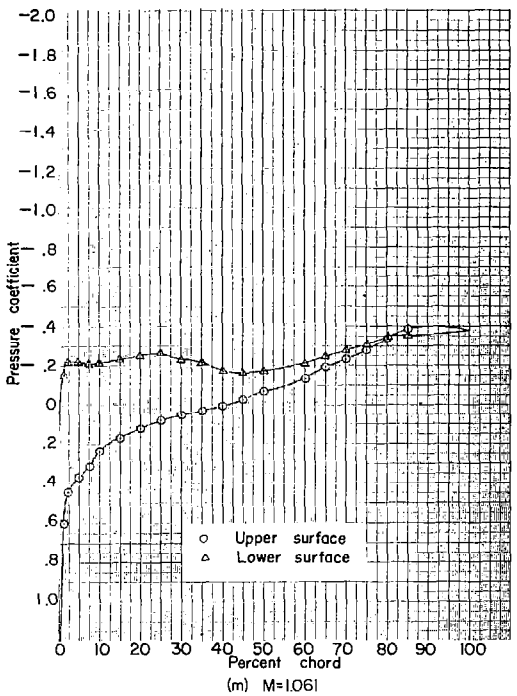


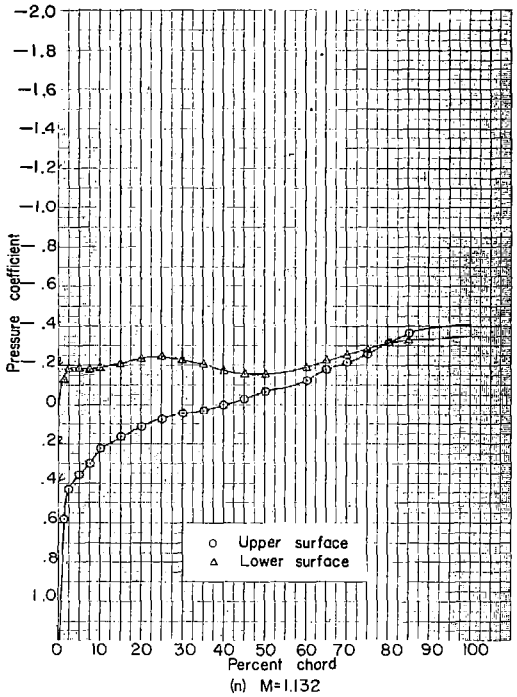
(k) M=1.010

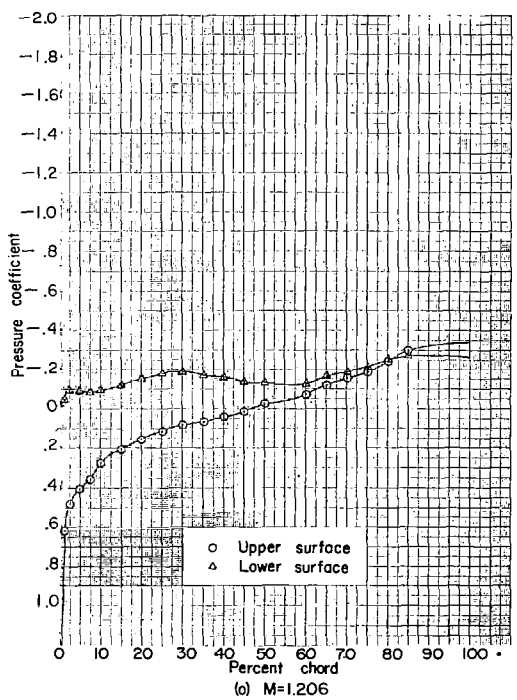


(l) M=1.035

Figure 37.- Continued. NACA 16-206; $\alpha = -2^{o}$.







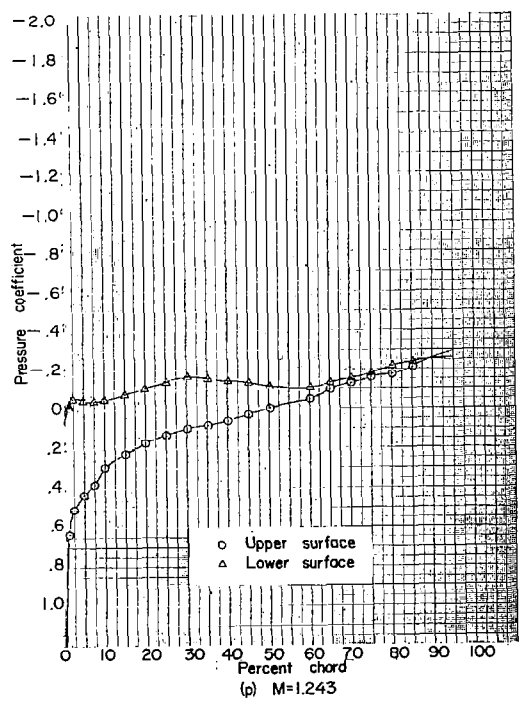


Figure 37.- Concluded. NACA 16-206; $\alpha = -2^{\circ}$.

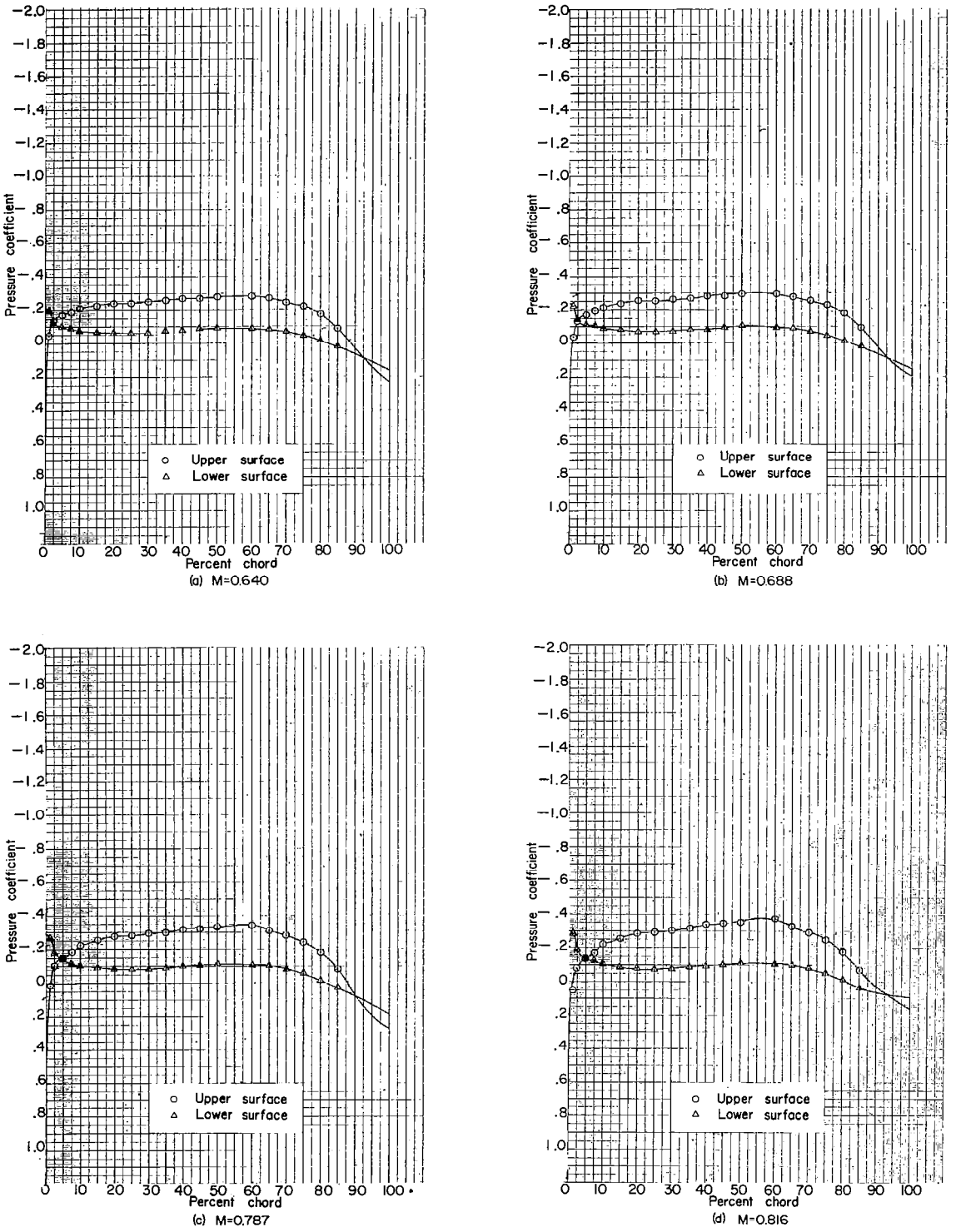
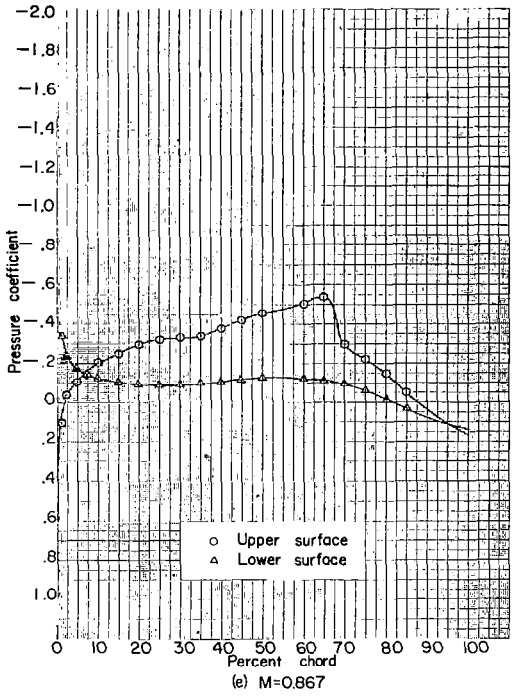
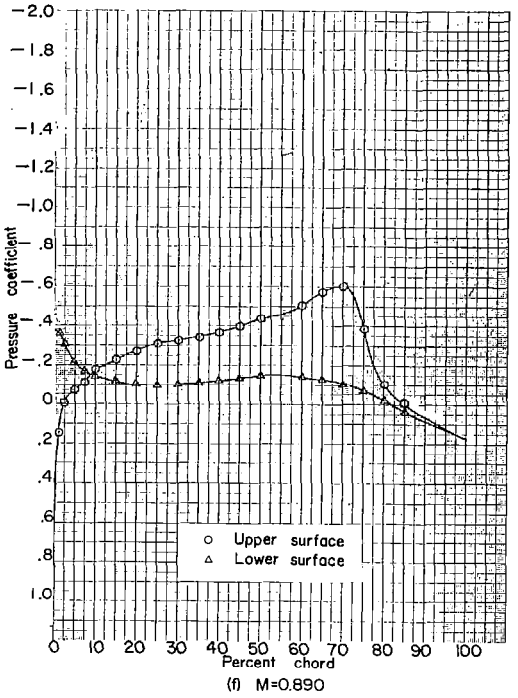


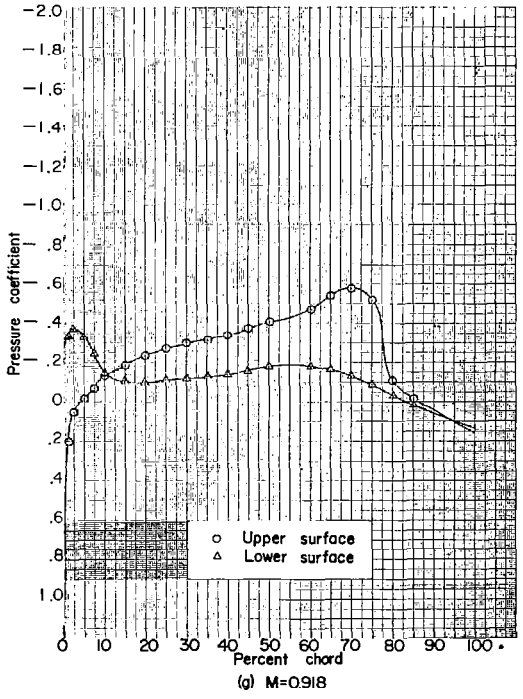
Figure 38.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 0^{\circ}$.



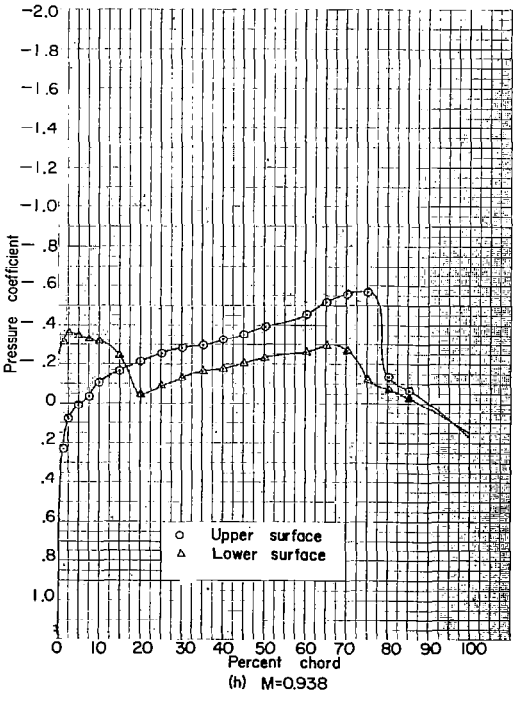
(e) M=0.867



(f) M=0.890

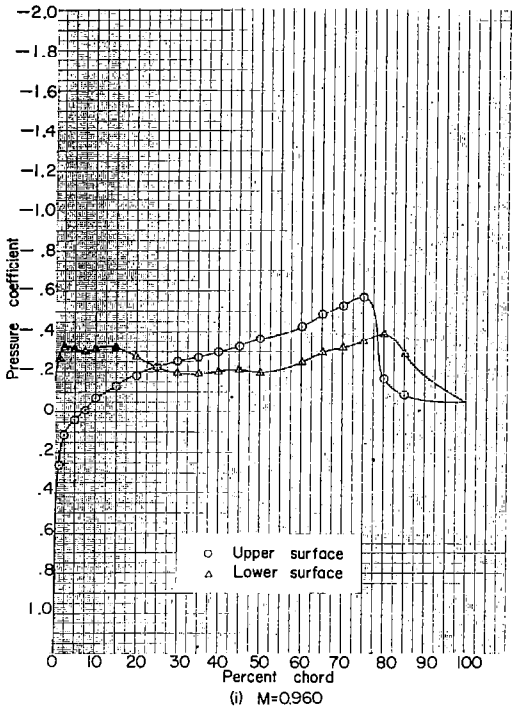


(g) M=0.918

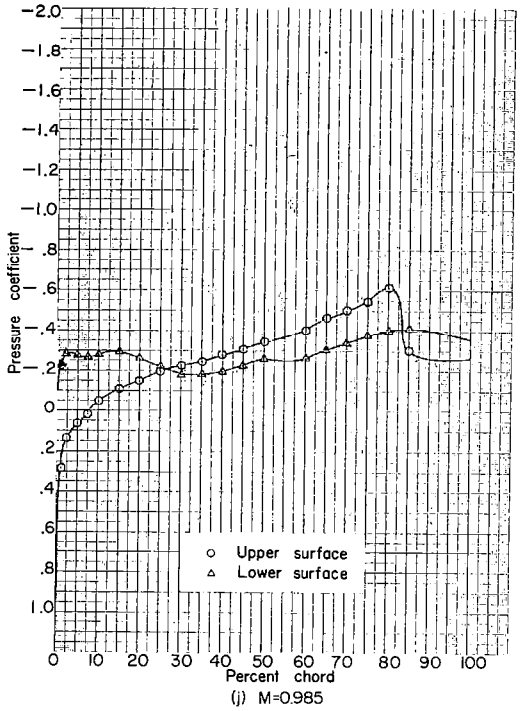


(h) M=0.938

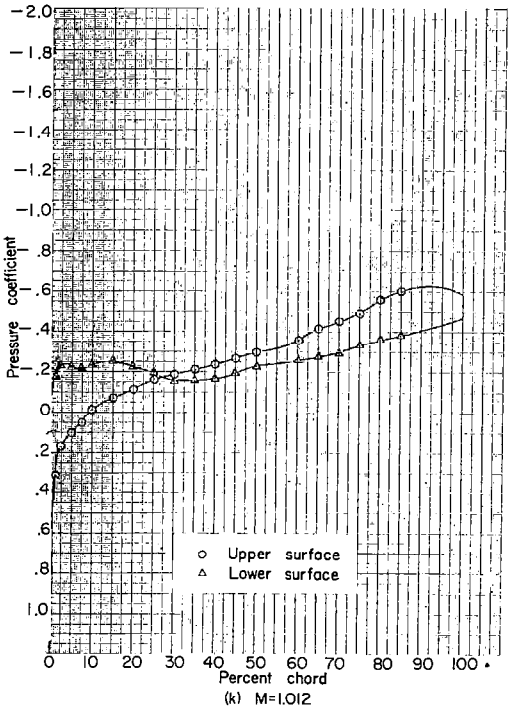
Figure 38.- Continued. NACA 16-206; $\alpha = 0^{\circ}$.



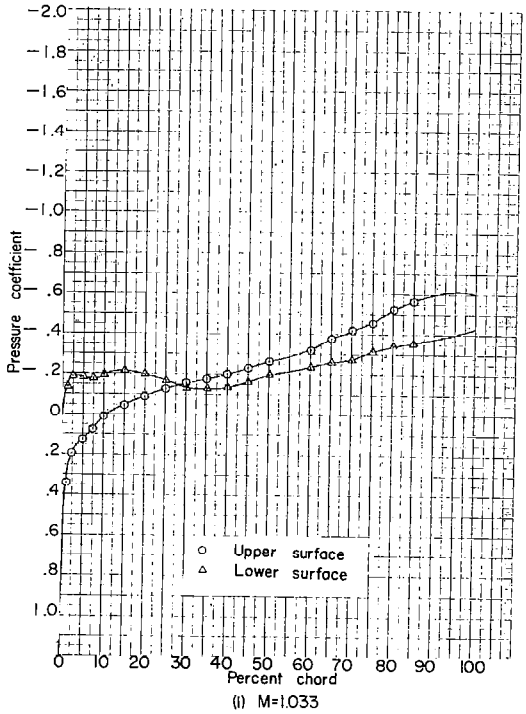
(i) M=0.960



(j) M=0.985

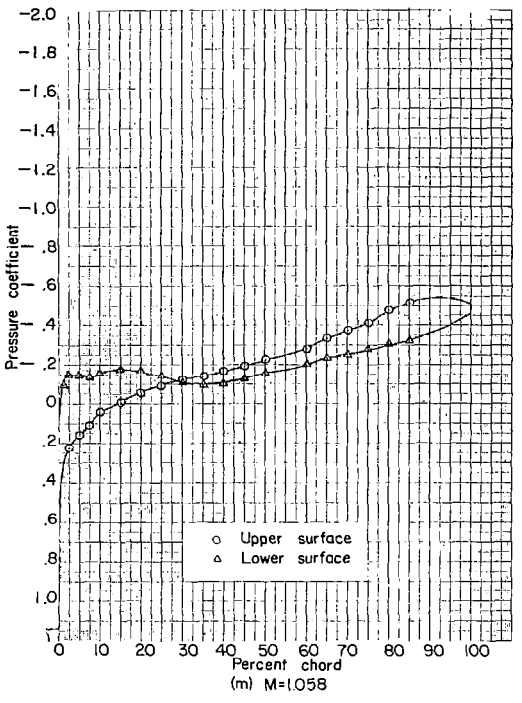


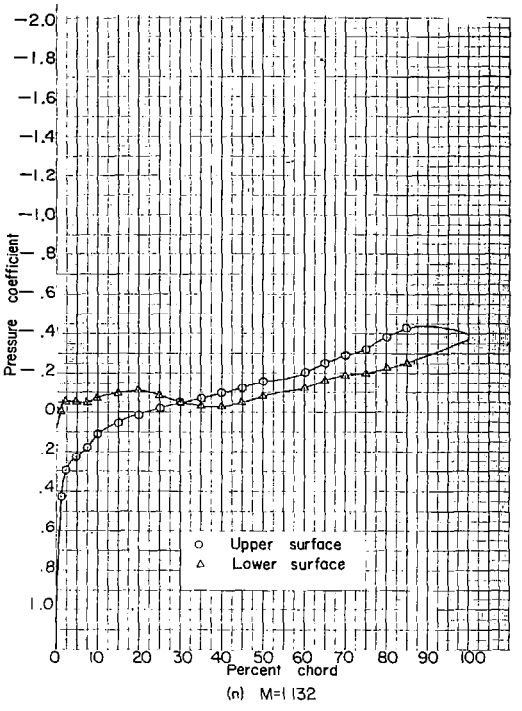
(k) M=1.012

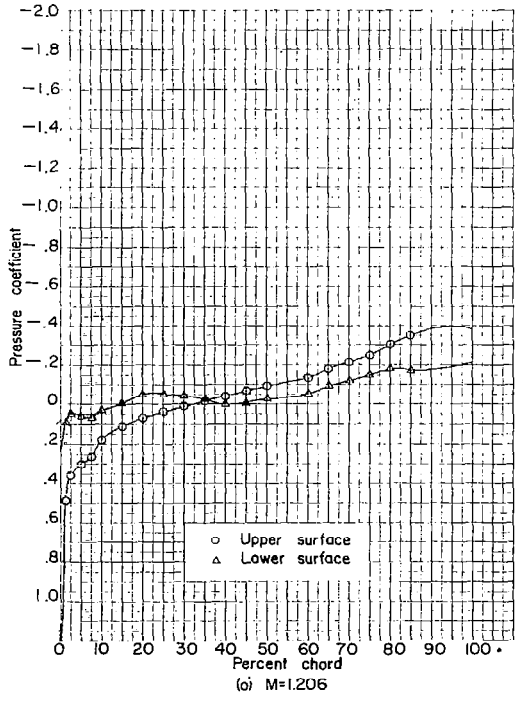


(l) M=1.033

Figure 38.- Continued. NACA 16-206; $\alpha = 0^{\circ}$.







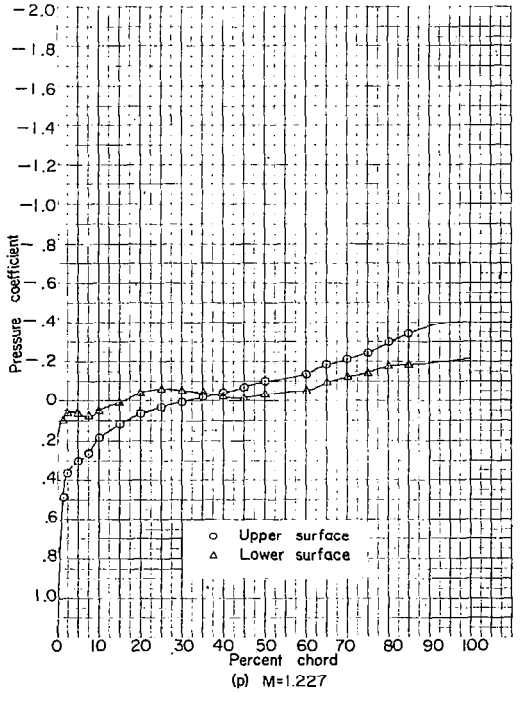


Figure 38.- Concluded. NACA 16-206; $\alpha = 0^{\circ}$.

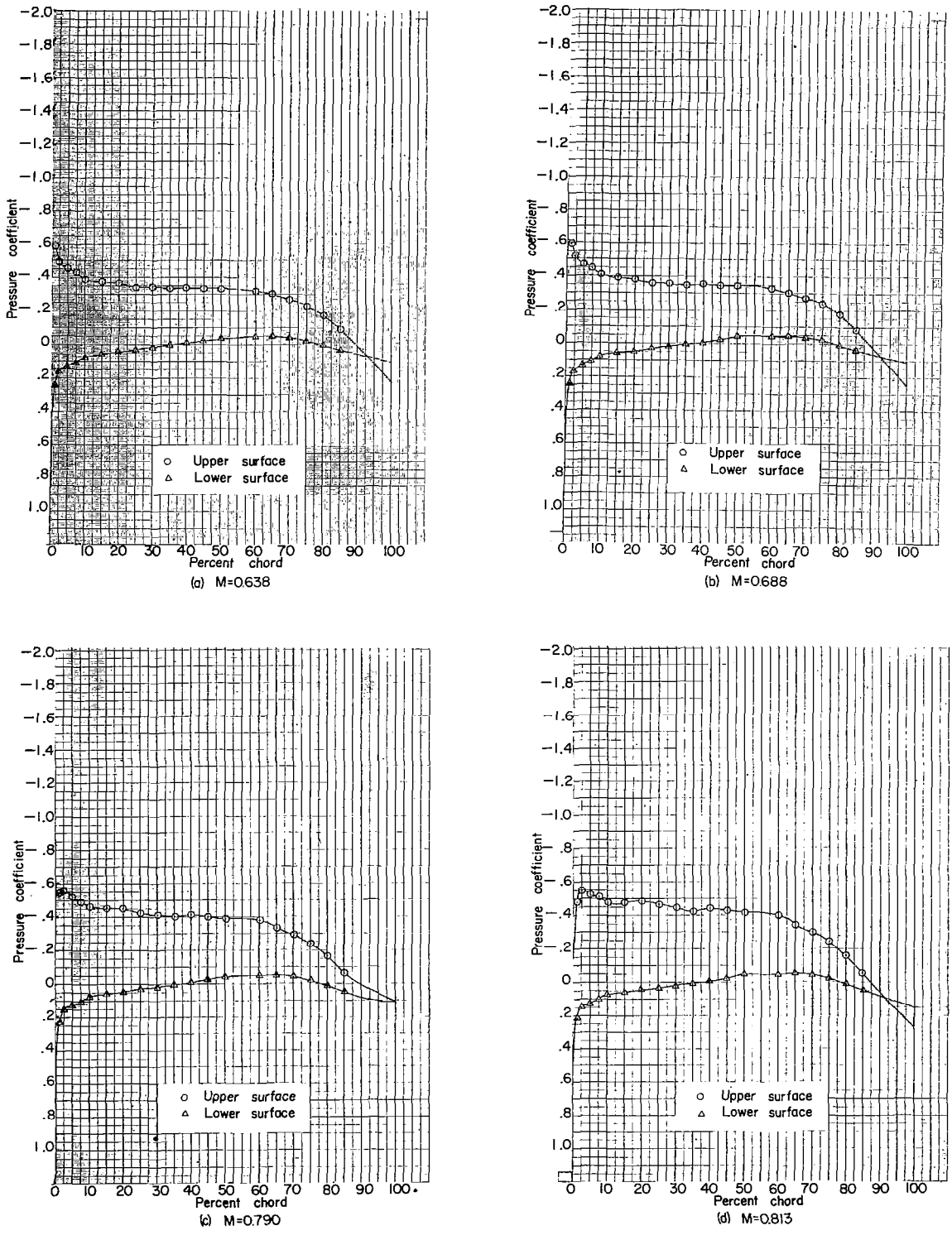
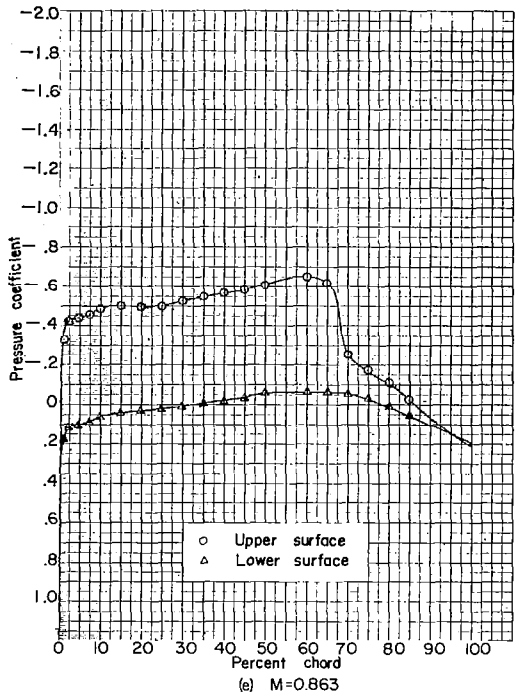
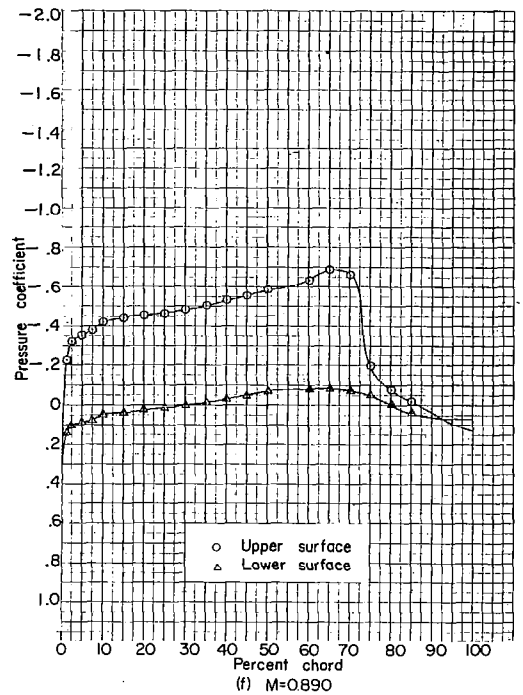
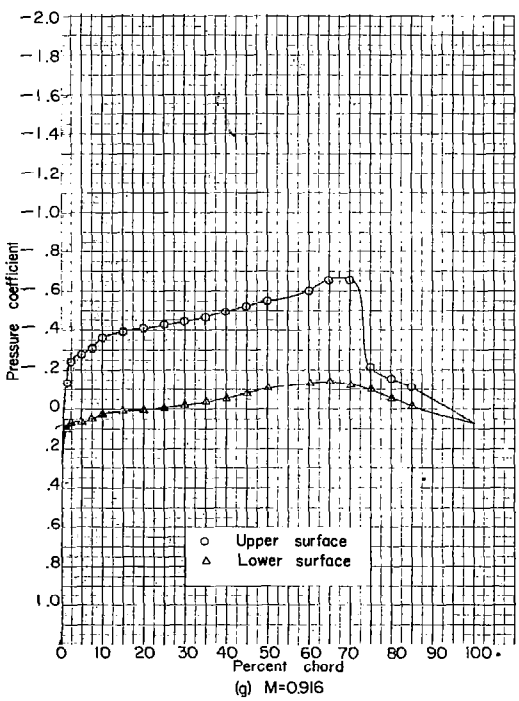


Figure 39.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 2^{\circ}$.







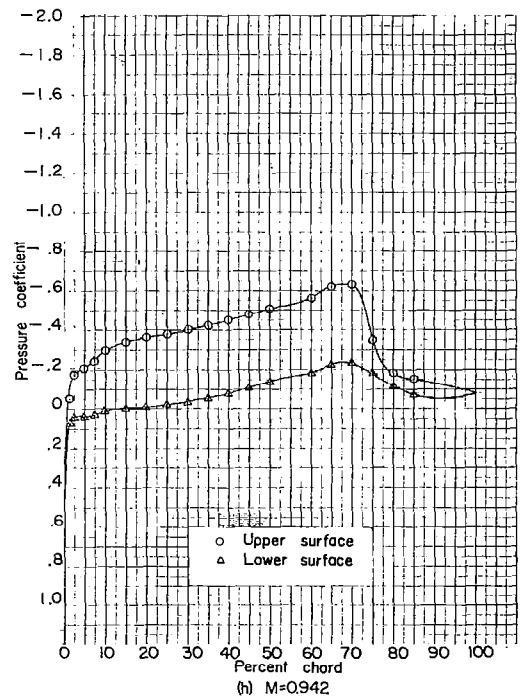
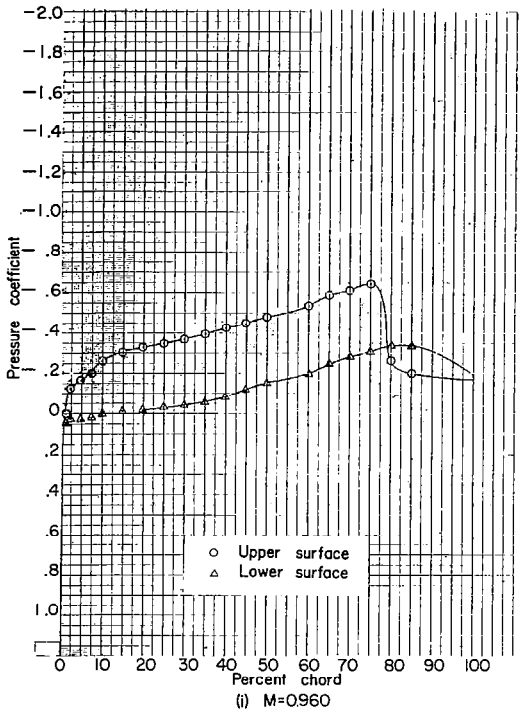
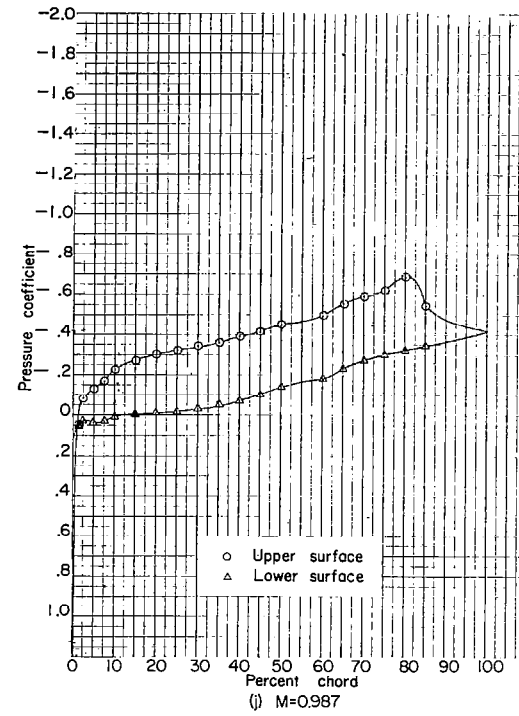
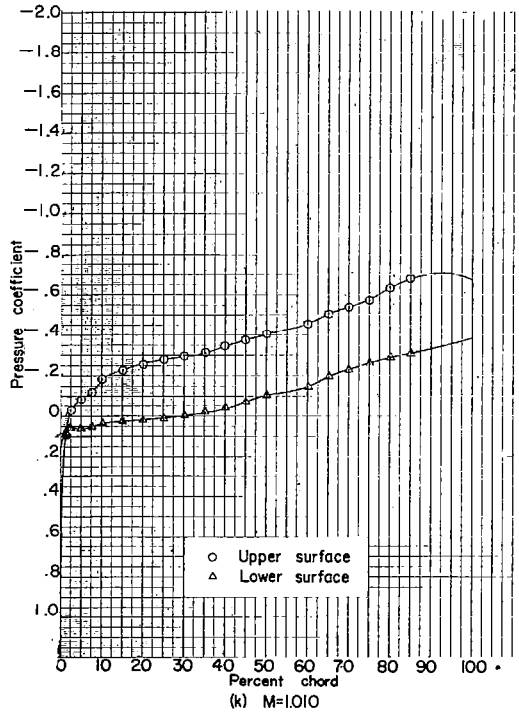


Figure 39.- Continued. NACA 16-206; $\alpha = 2^{\circ}$.







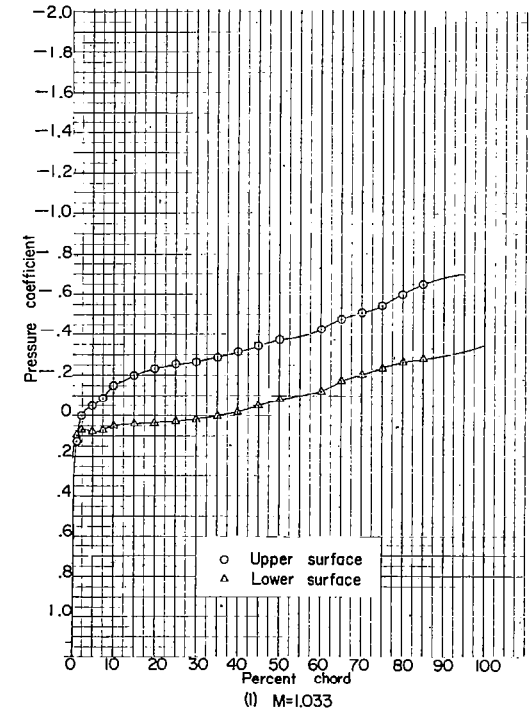


Figure 39.- Continued. NACA 16-206; $\alpha = 2^{\circ}$.

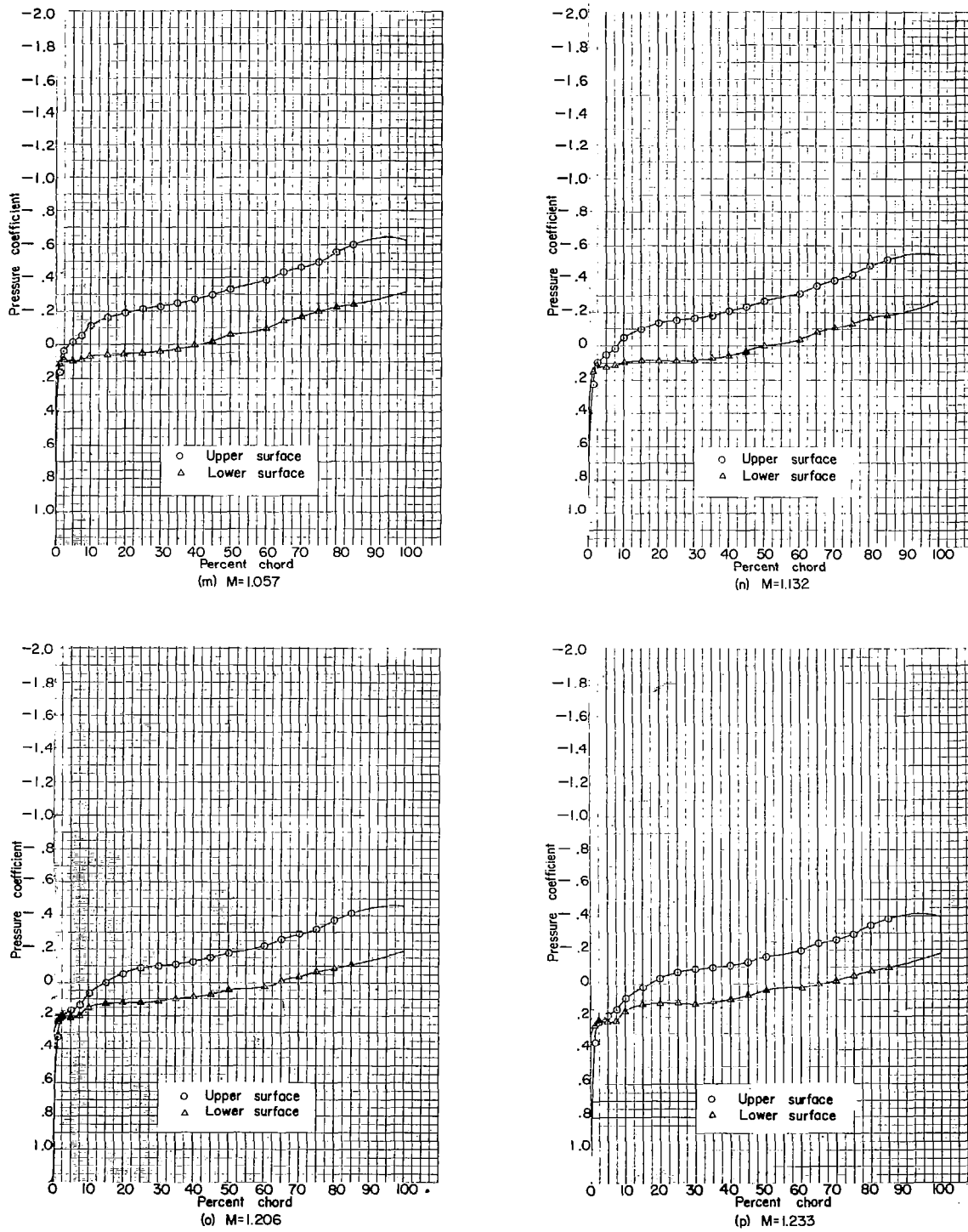


Figure 39.- Concluded. NACA 16-206; $\alpha = 2^{\circ}$.

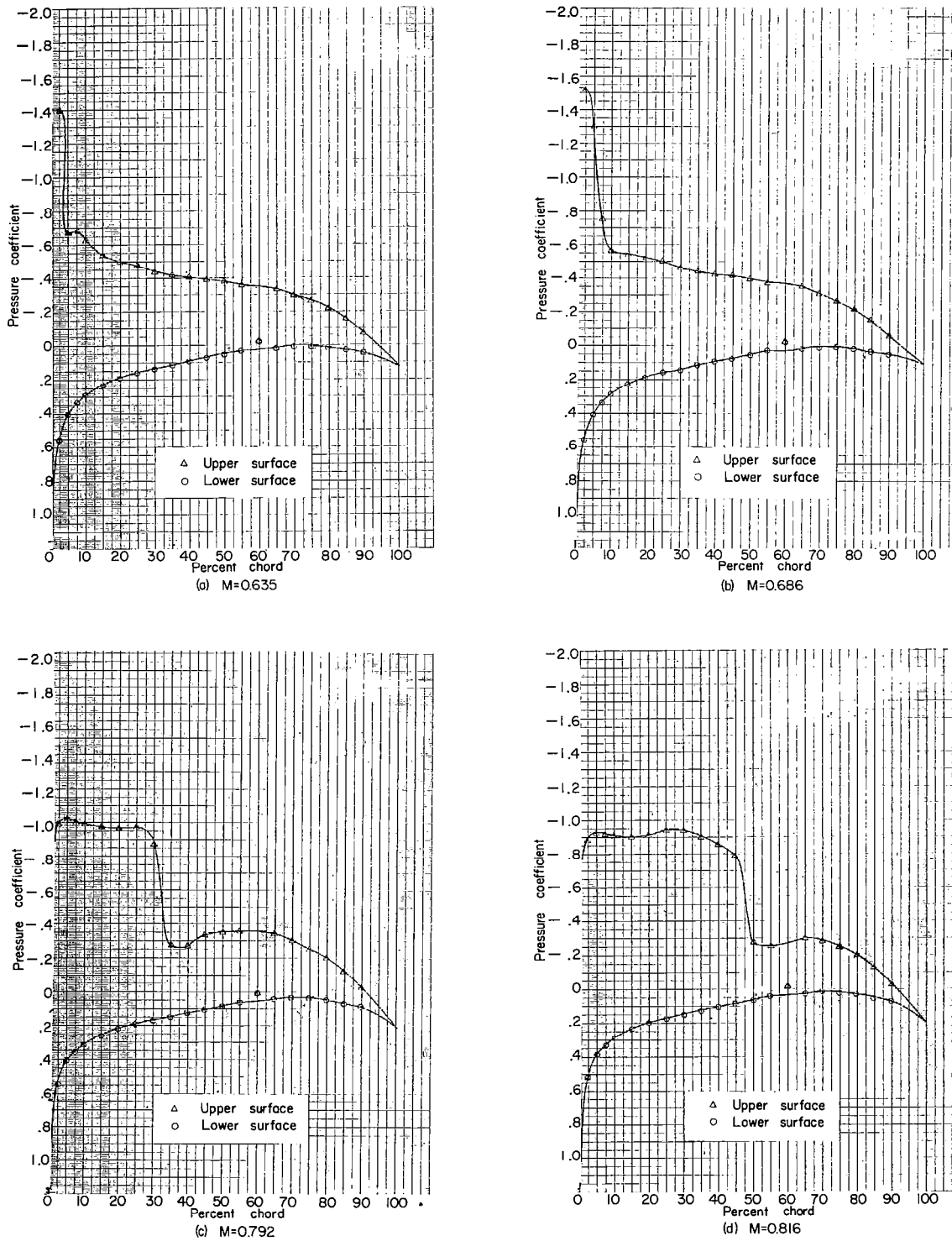
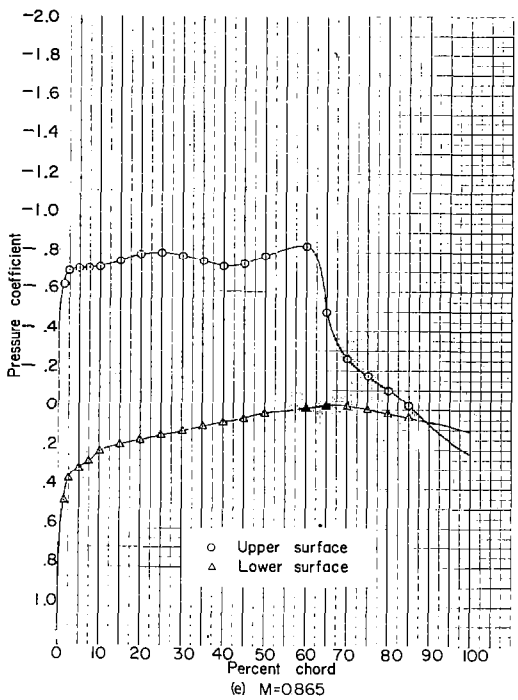
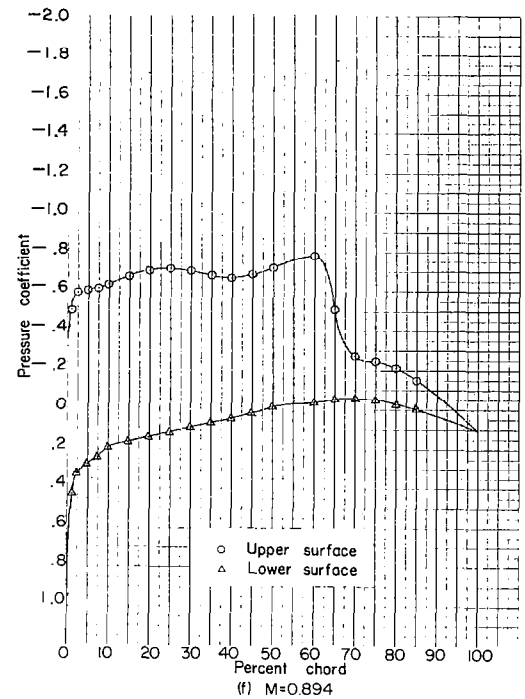


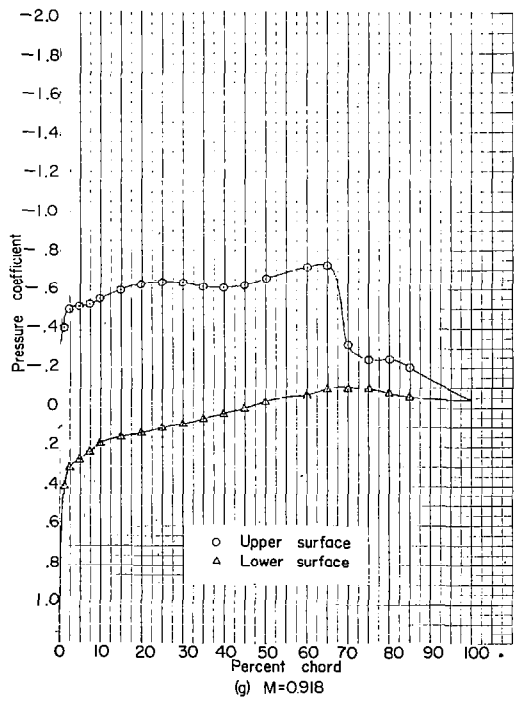
Figure 40.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 4^{\circ}$.



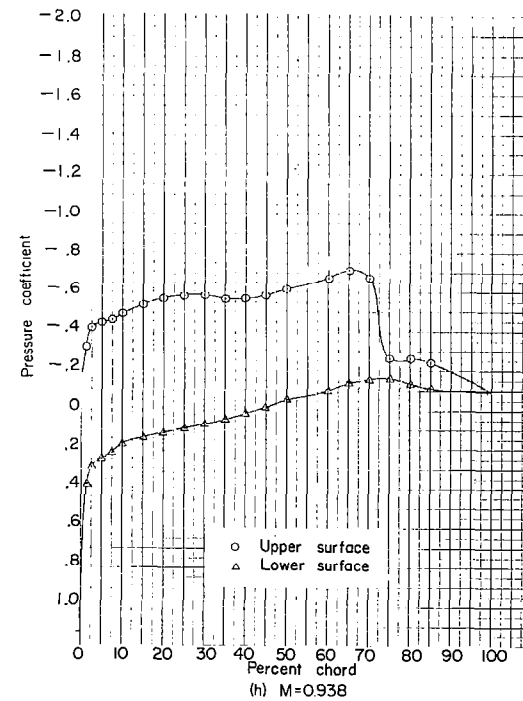
(e) M=0.865



(f) M=0.894

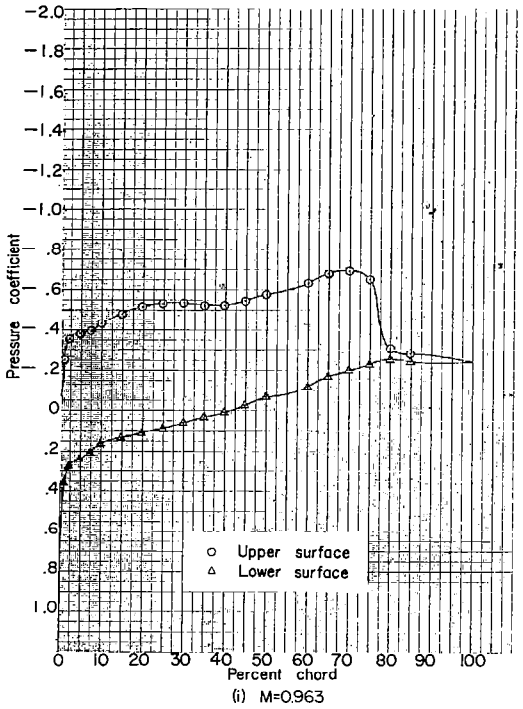


(g) M=0.918

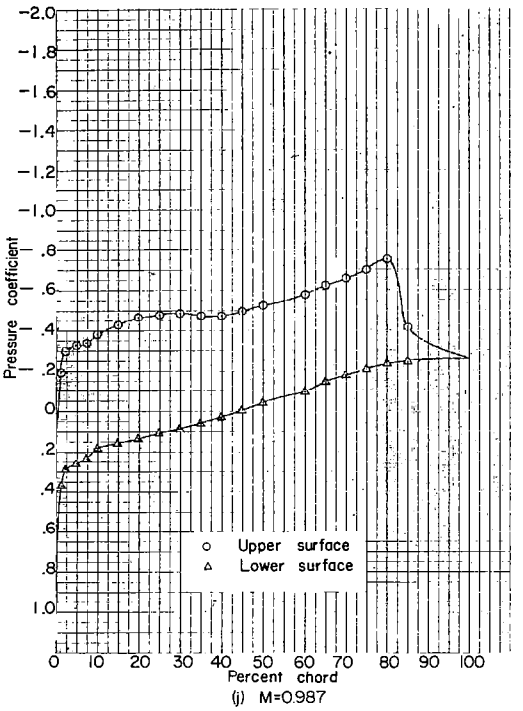


(h) M=0.938

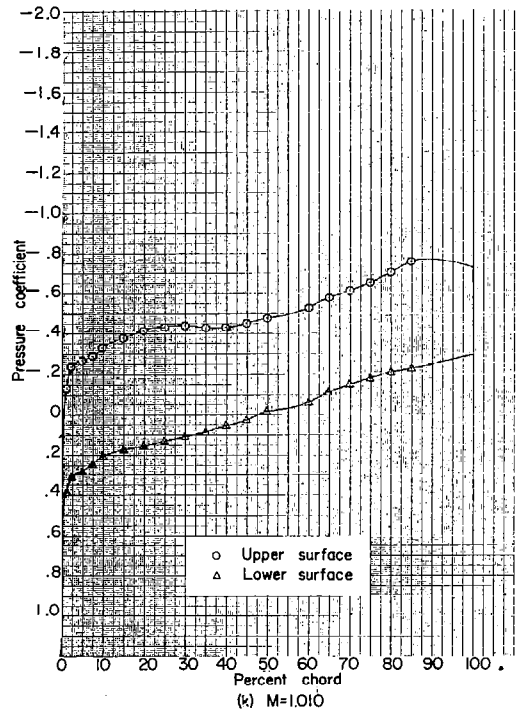
Figure 40.- Continued. NACA 16-206; $\alpha = 4^{\circ}$.



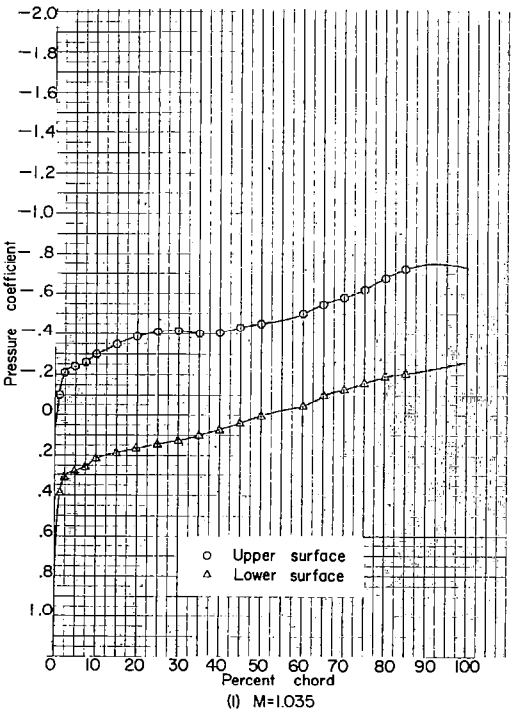
(i) M=0.963



(j) M=0.987

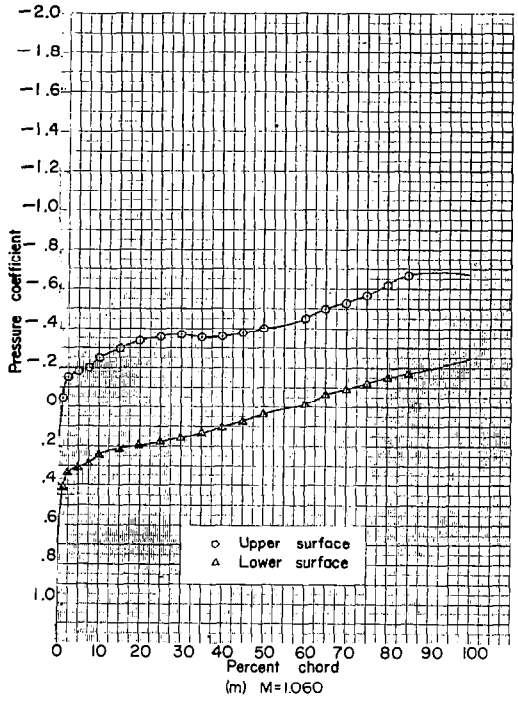


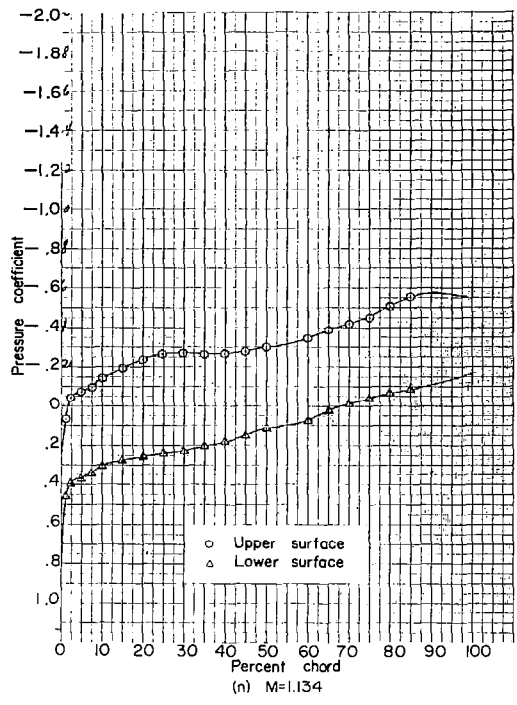
(k) M=1.010

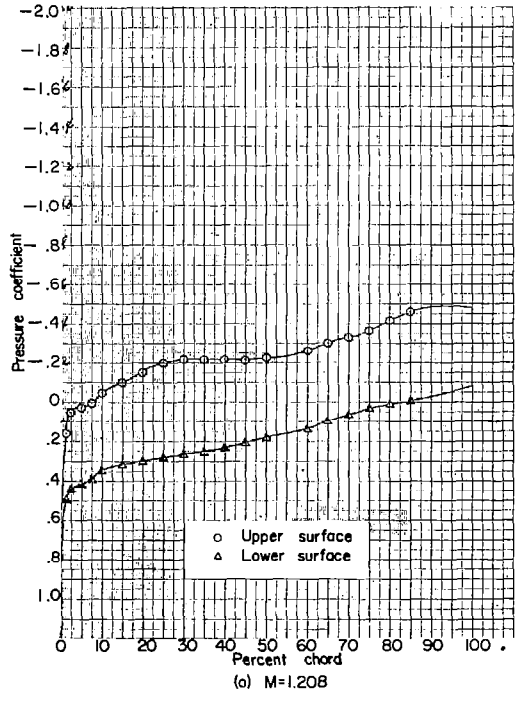


(l) M=1.035

Figure 40.- Continued. NACA 16-206; $\alpha = 4^{\circ}$.







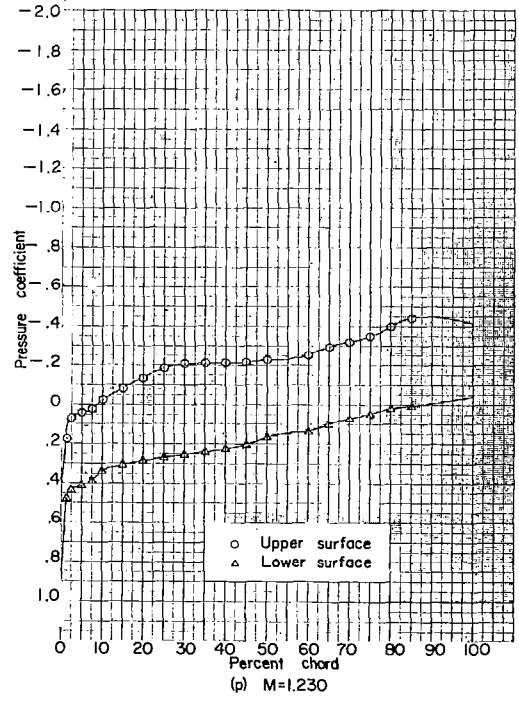


Figure 40.- Concluded. NACA 16-206; $\alpha = 4^{\circ}$.

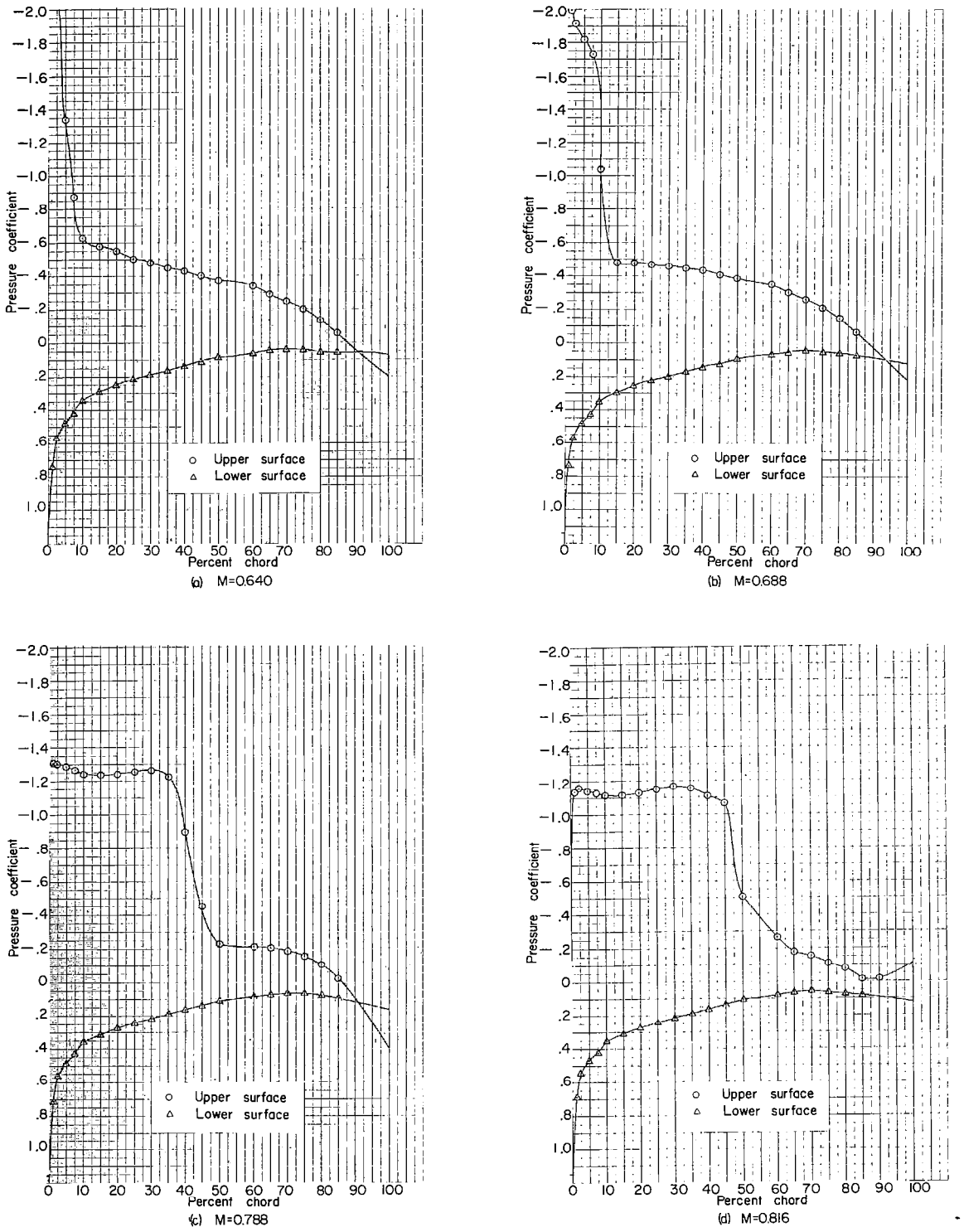
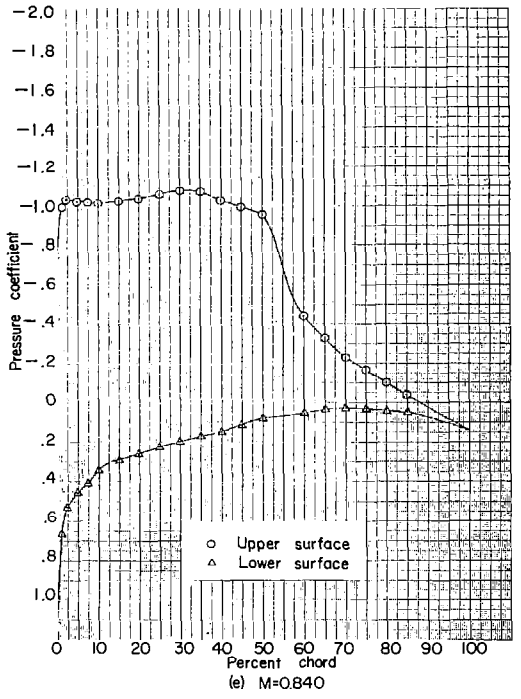
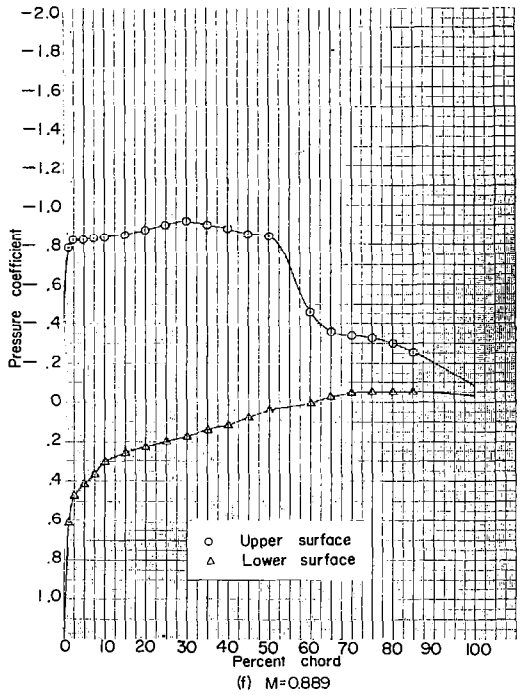


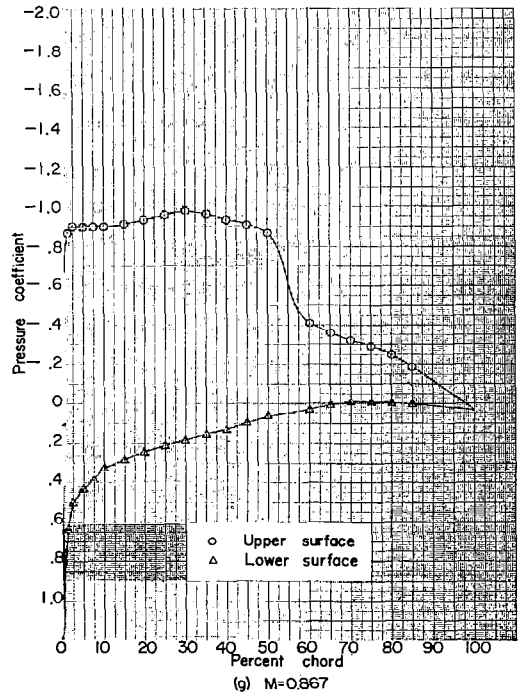
Figure 41.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 6^{\circ}$.



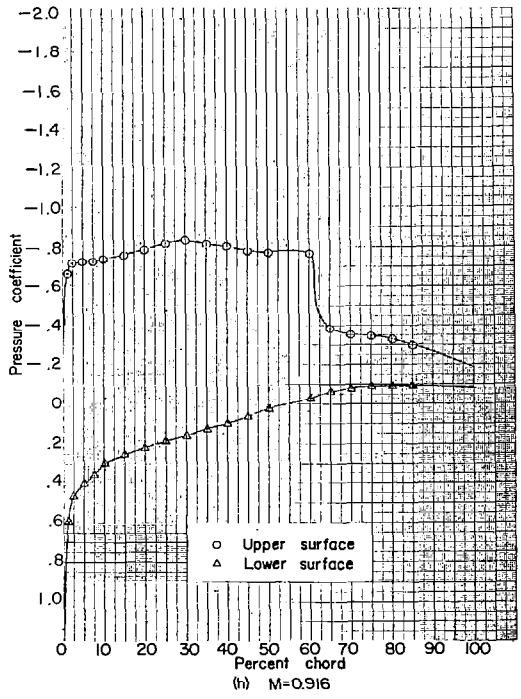
(e) M=0.840



(f) M=0.889

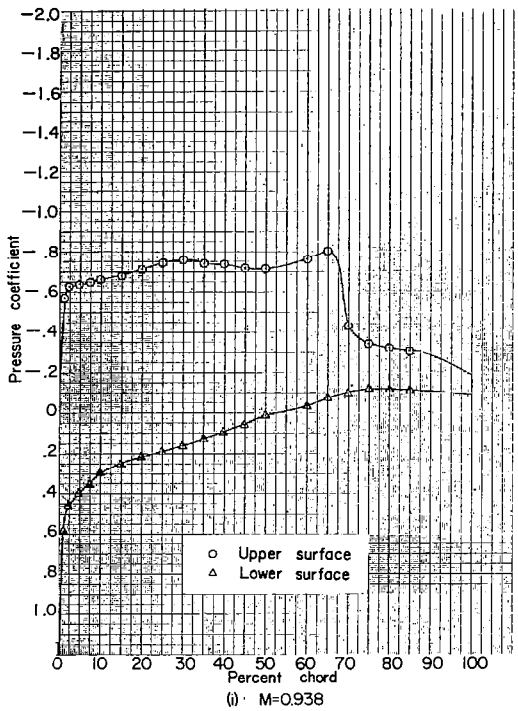


(g) M=0.867

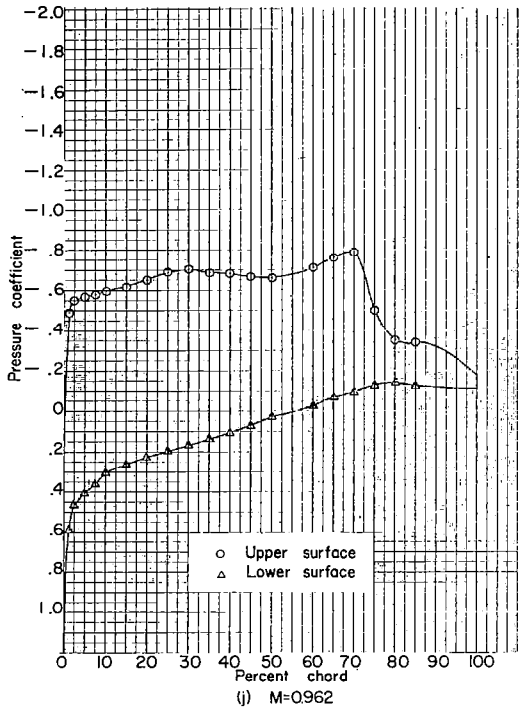


(h) M=0.916

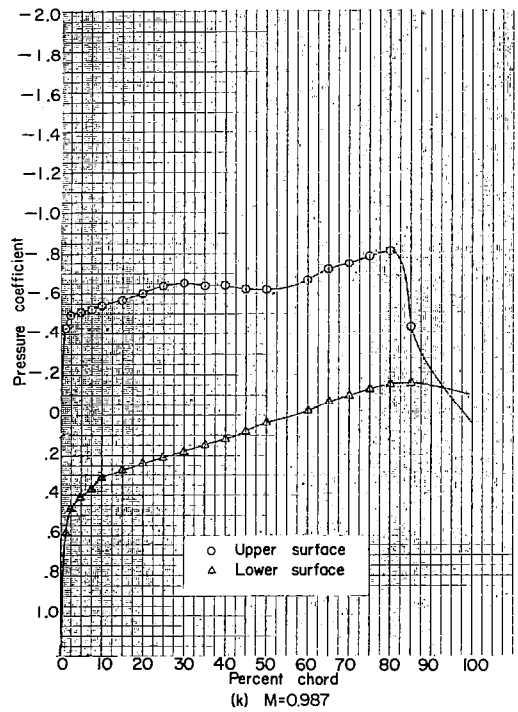
Figure 41.- Continued. NACA 16-206; $\alpha = 6^{\circ}$.



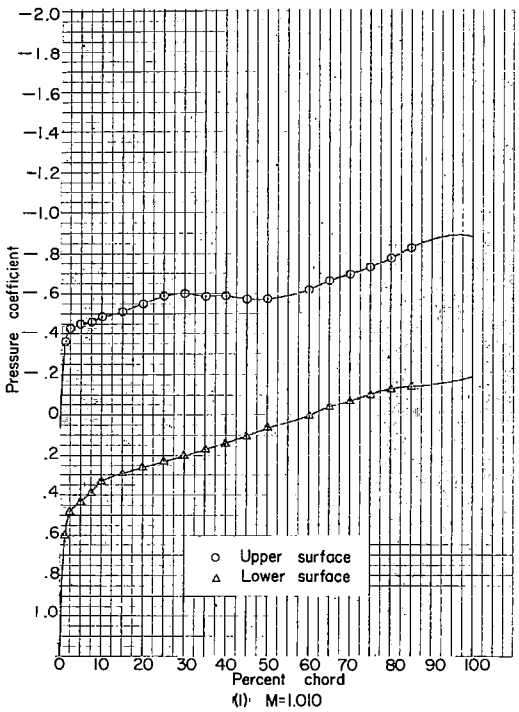
(i) M=0.938



(j) M=0.962

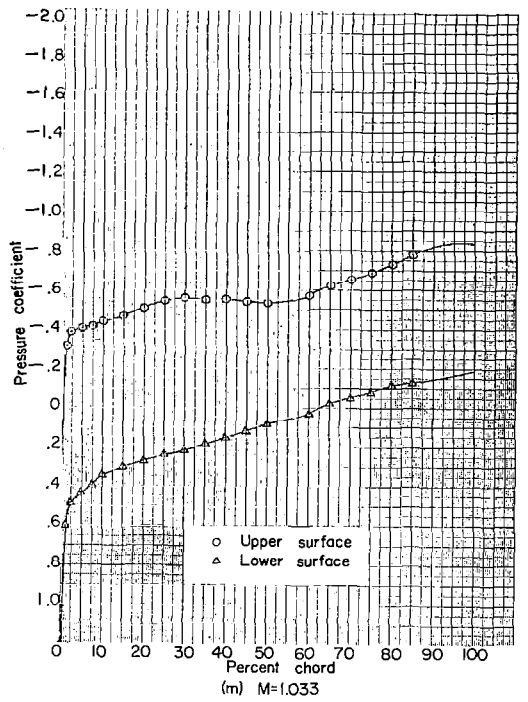


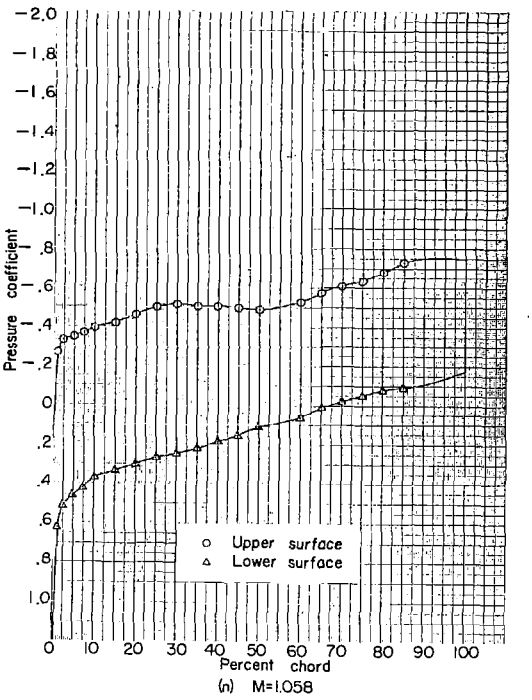
(k) M=0.987

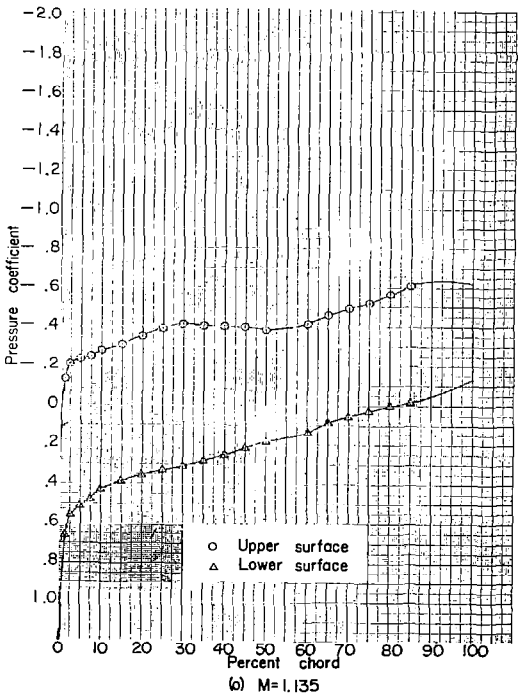


(l) M=1.010

Figure 41.- Continued. NACA 16-206; $\alpha = 6^{\circ}$.







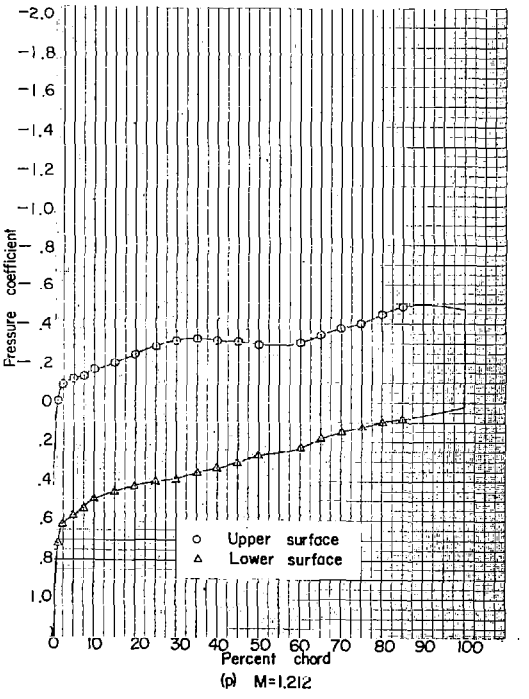


Figure 41.- Concluded. NACA 16-206; $\alpha = 6^{\circ}$.

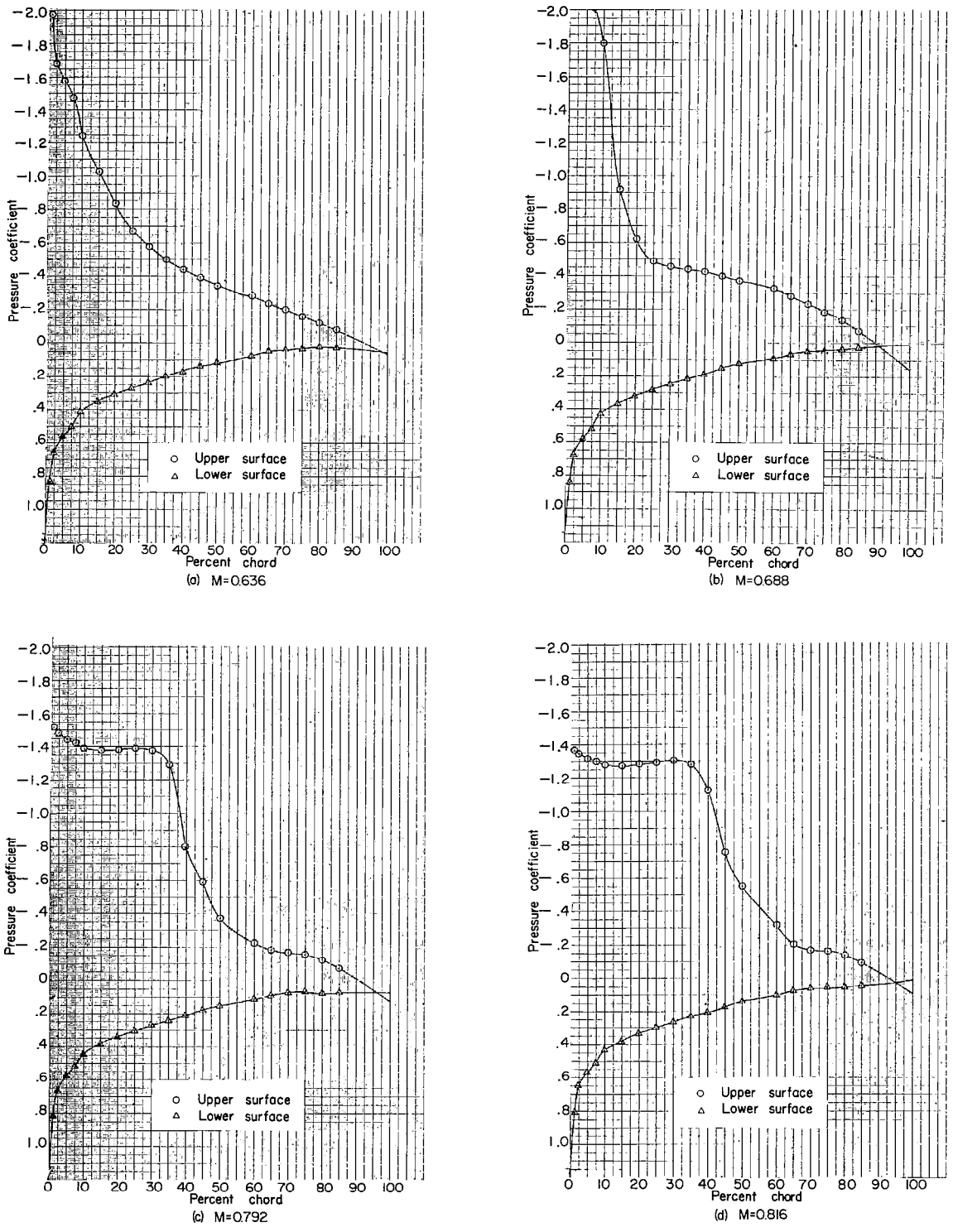
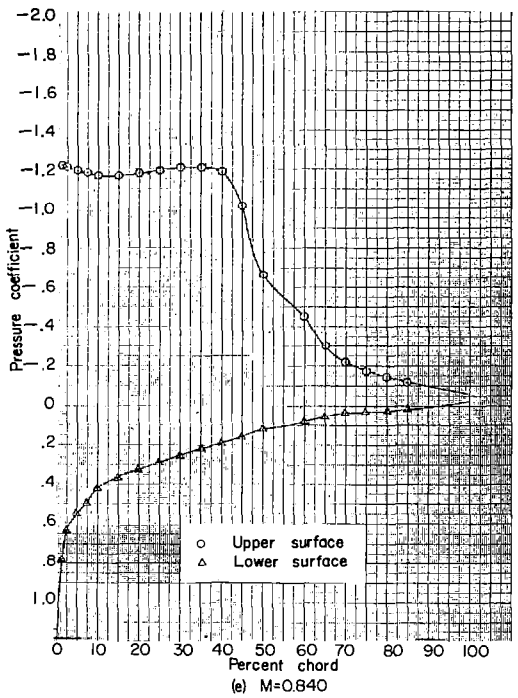
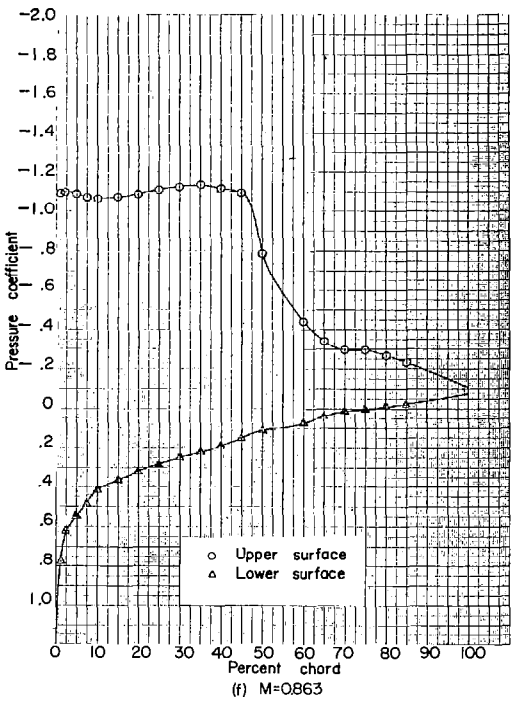
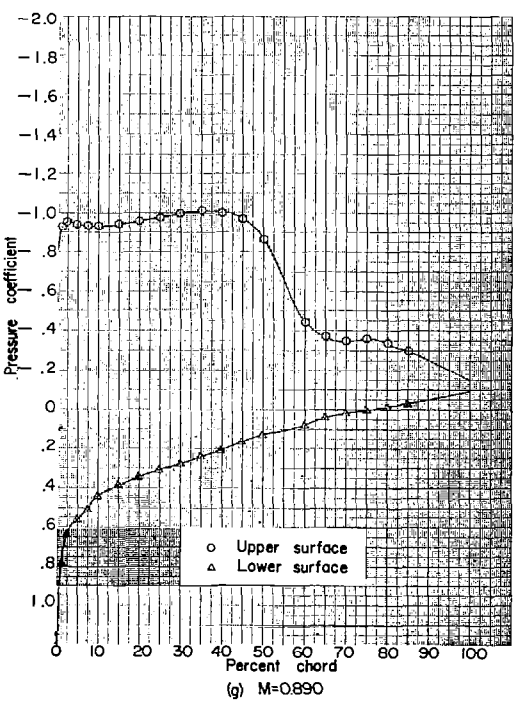


Figure 42.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 8^{\circ}$.







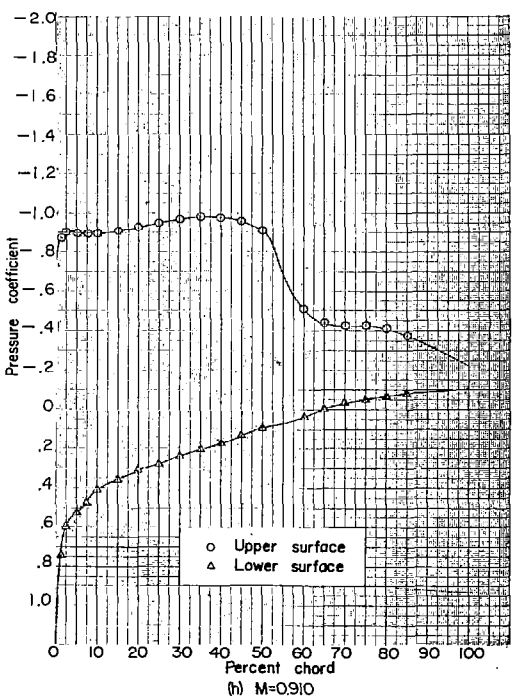
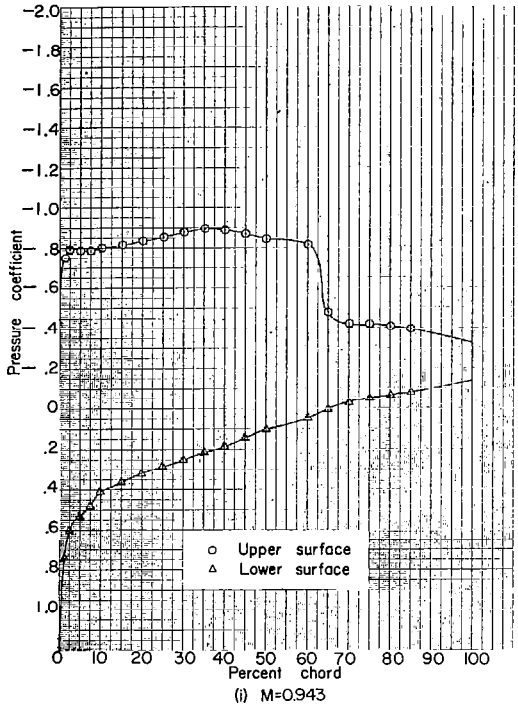
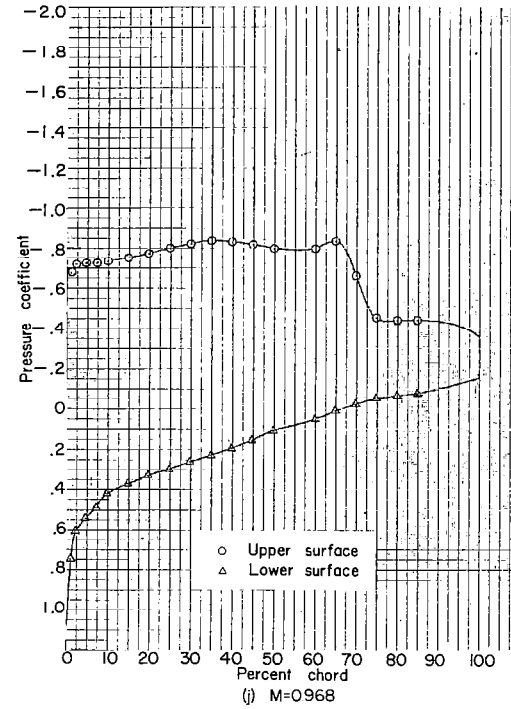
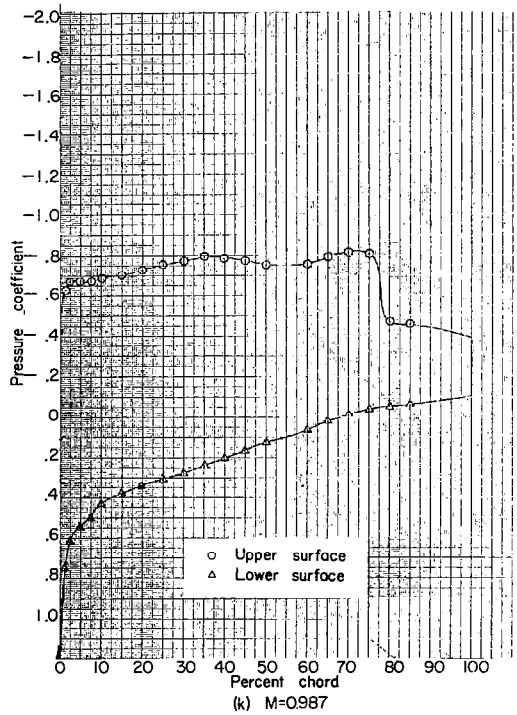


Figure 42.- Continued. NACA 16-206; $\alpha = 8^{\circ}$.







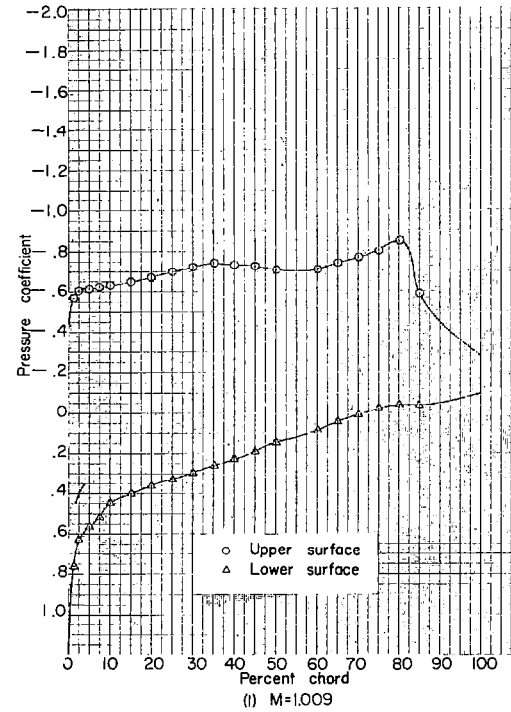
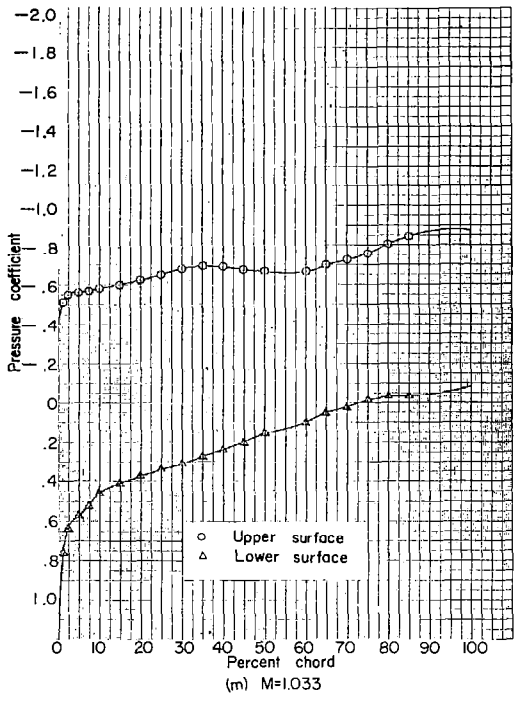
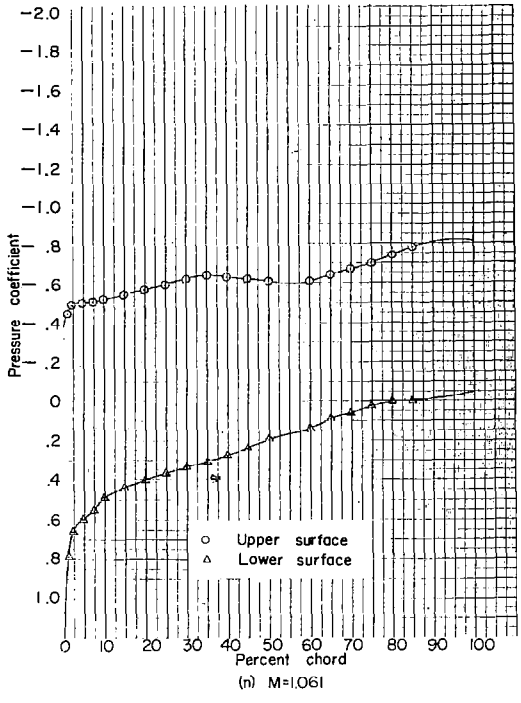


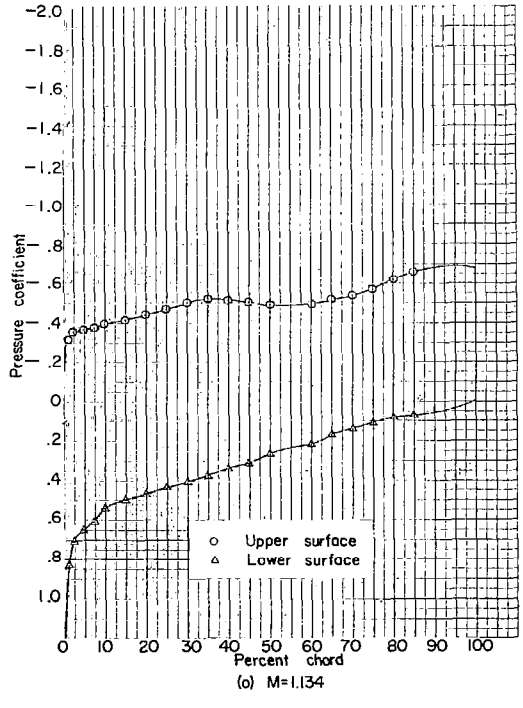
Figure 42.- Continued. NACA 16-206; $\alpha = 8°$.



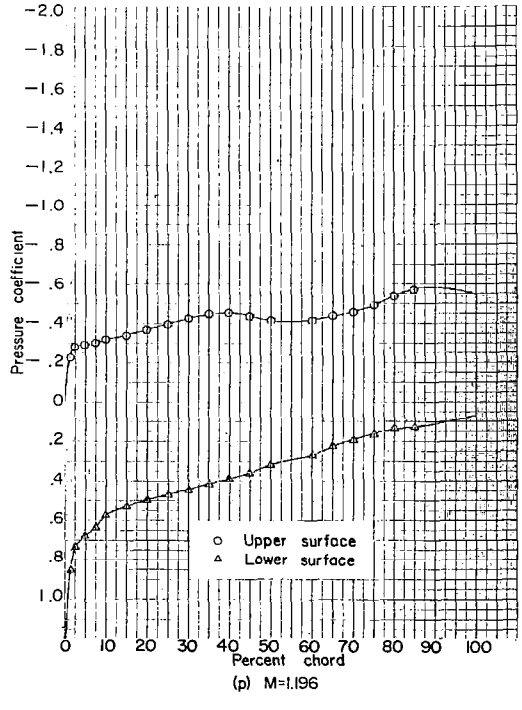
(m) M=1.033



(n) M=1.061



(o) M=1.134



(p) M=1.196

Figure 42.- Concluded. NACA 16-206; $\alpha = 8^{\circ}$.

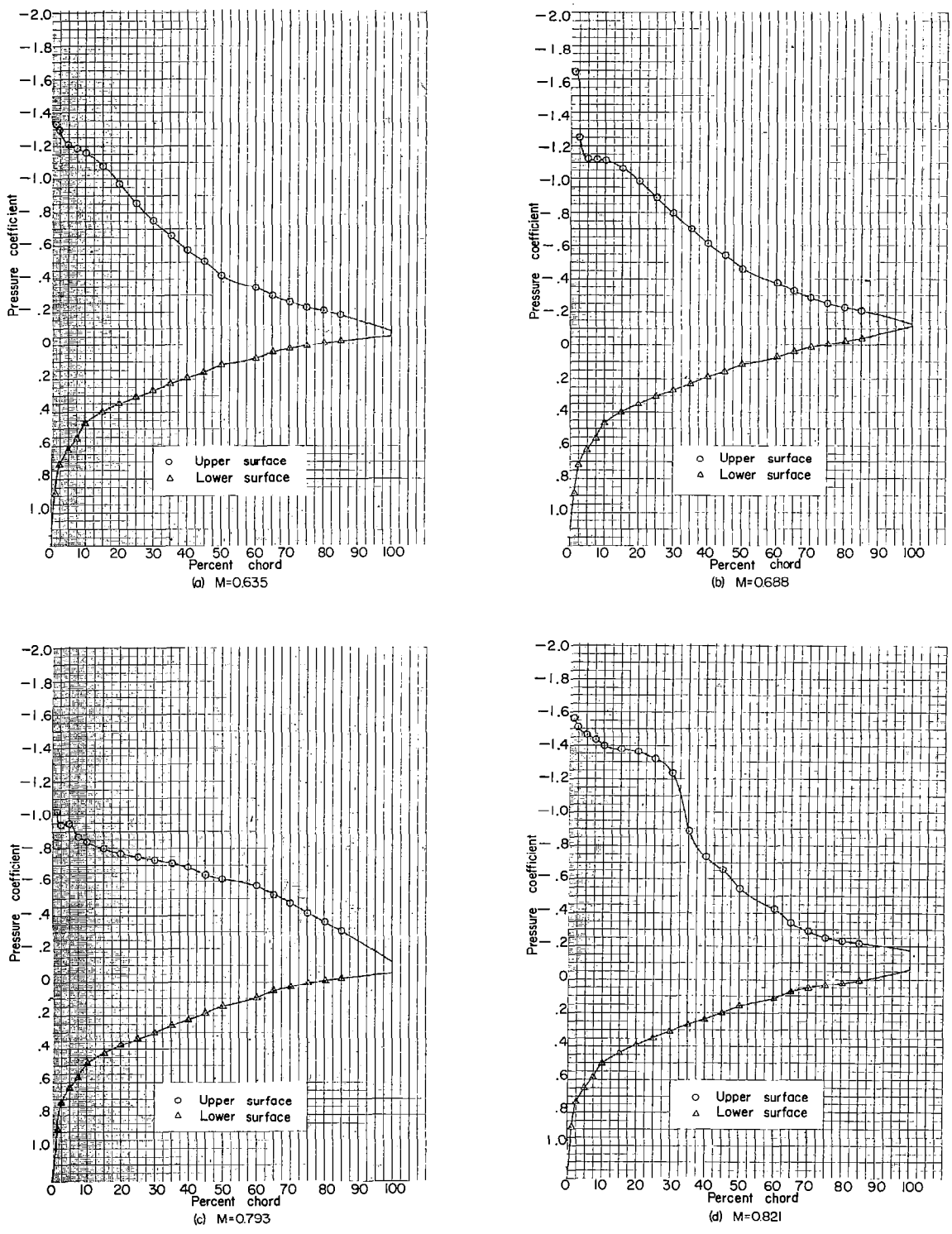
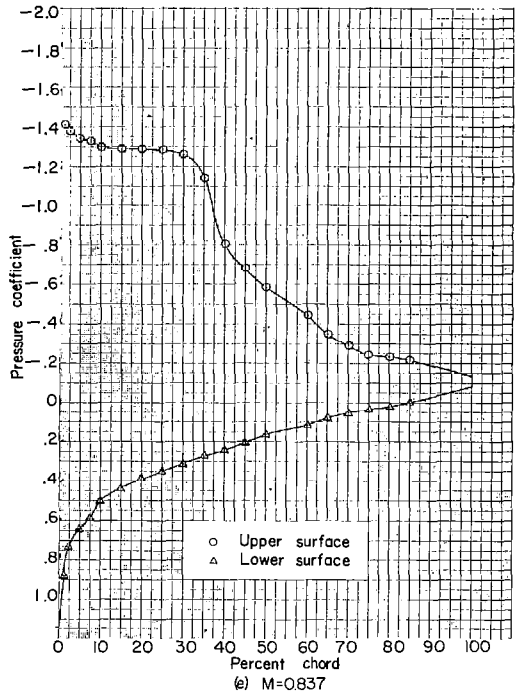
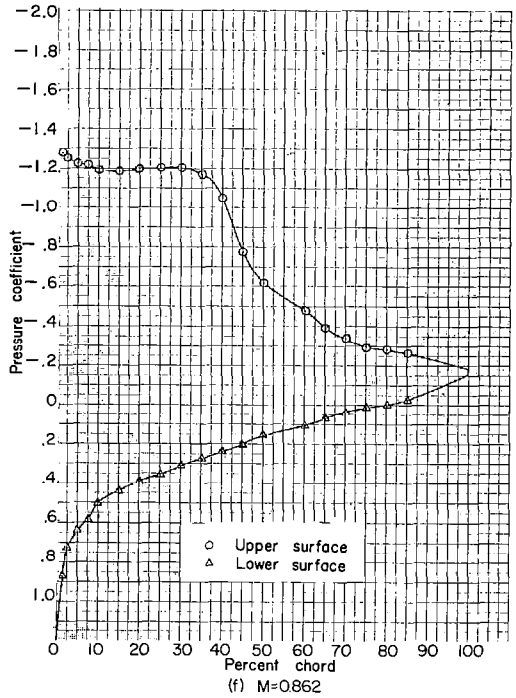


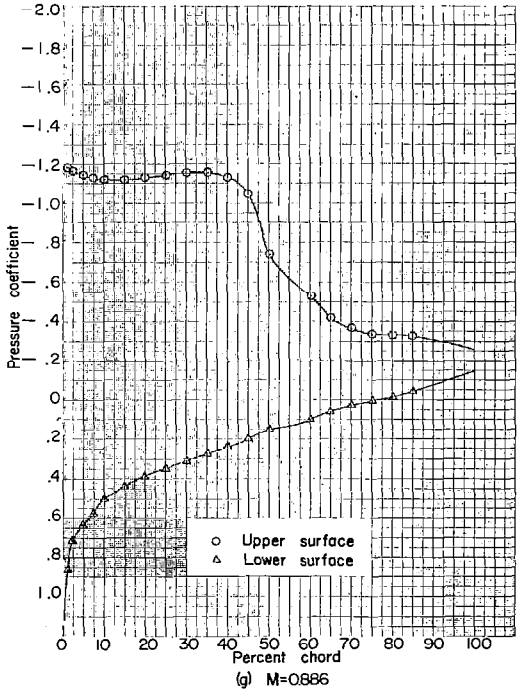
Figure 43.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 10^{\circ}$.



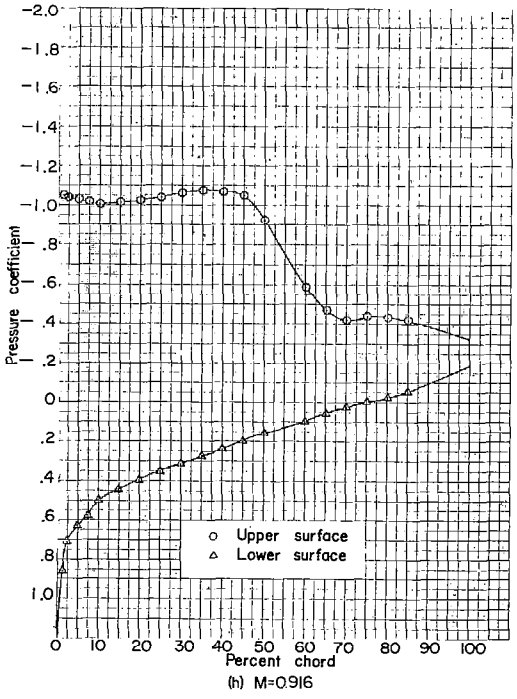
(e) M=0.837



(f) M=0.862

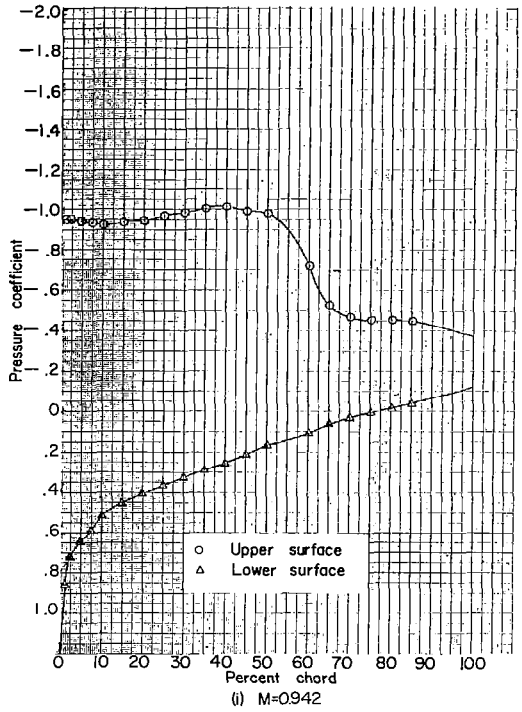


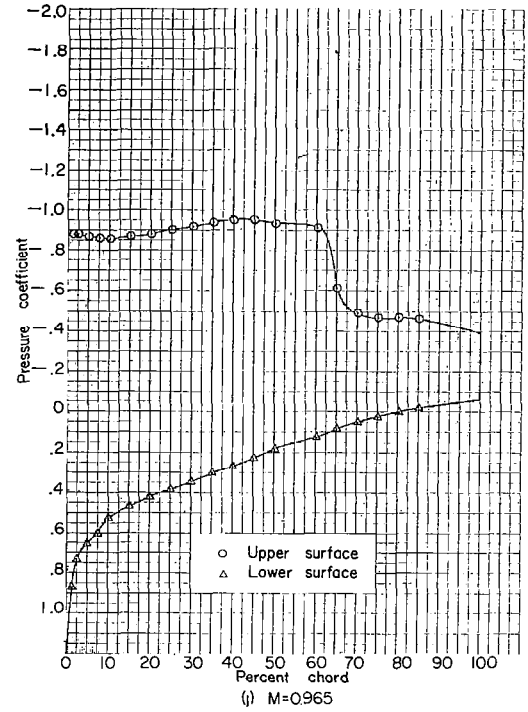
(g) M=0.886

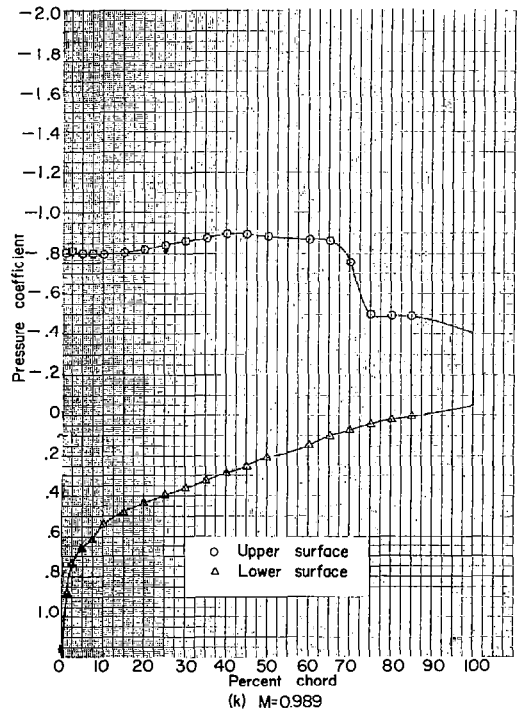


(h) M=0.916

Figure 43.- Continued. NACA 16-206; $\alpha = 10^{\circ}$.







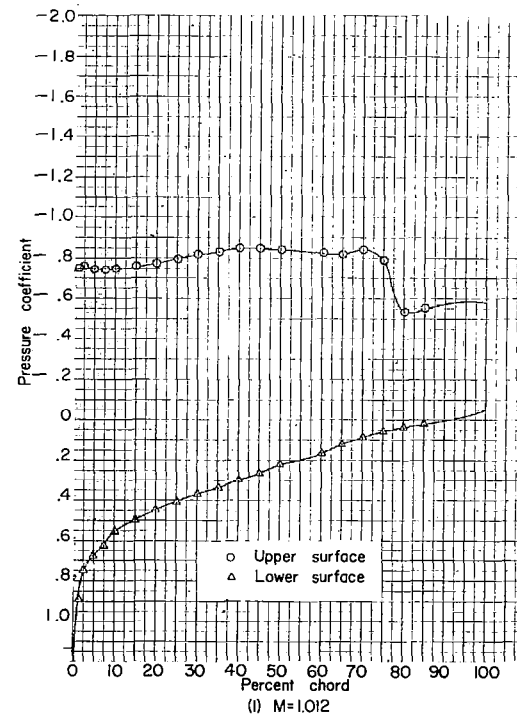


Figure 43.- Continued. NACA 16-206; $\alpha = 10^{\circ}$.

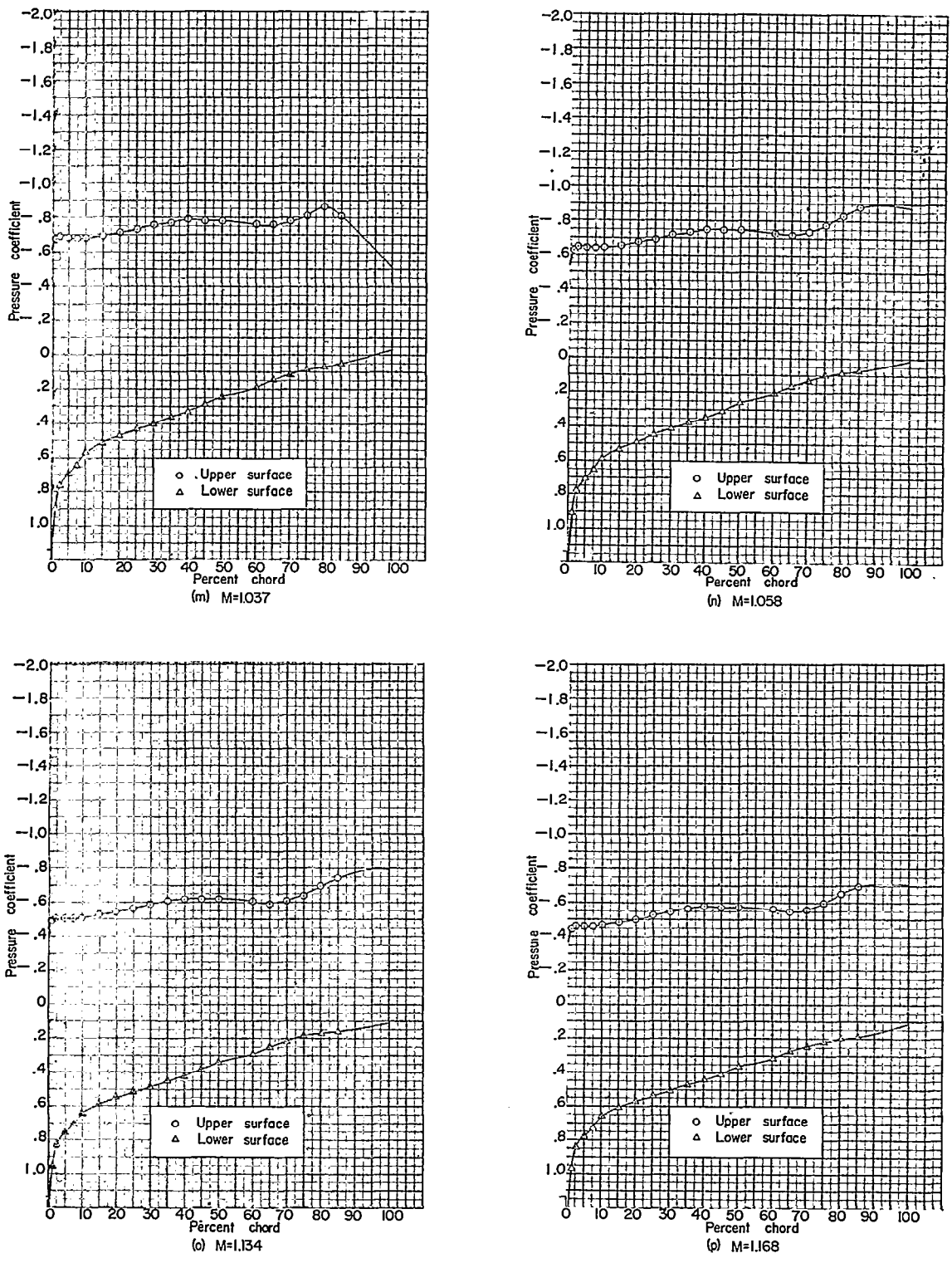


Figure 43.- Concluded. NACA 16-206; $\alpha = 10^{\circ}$.

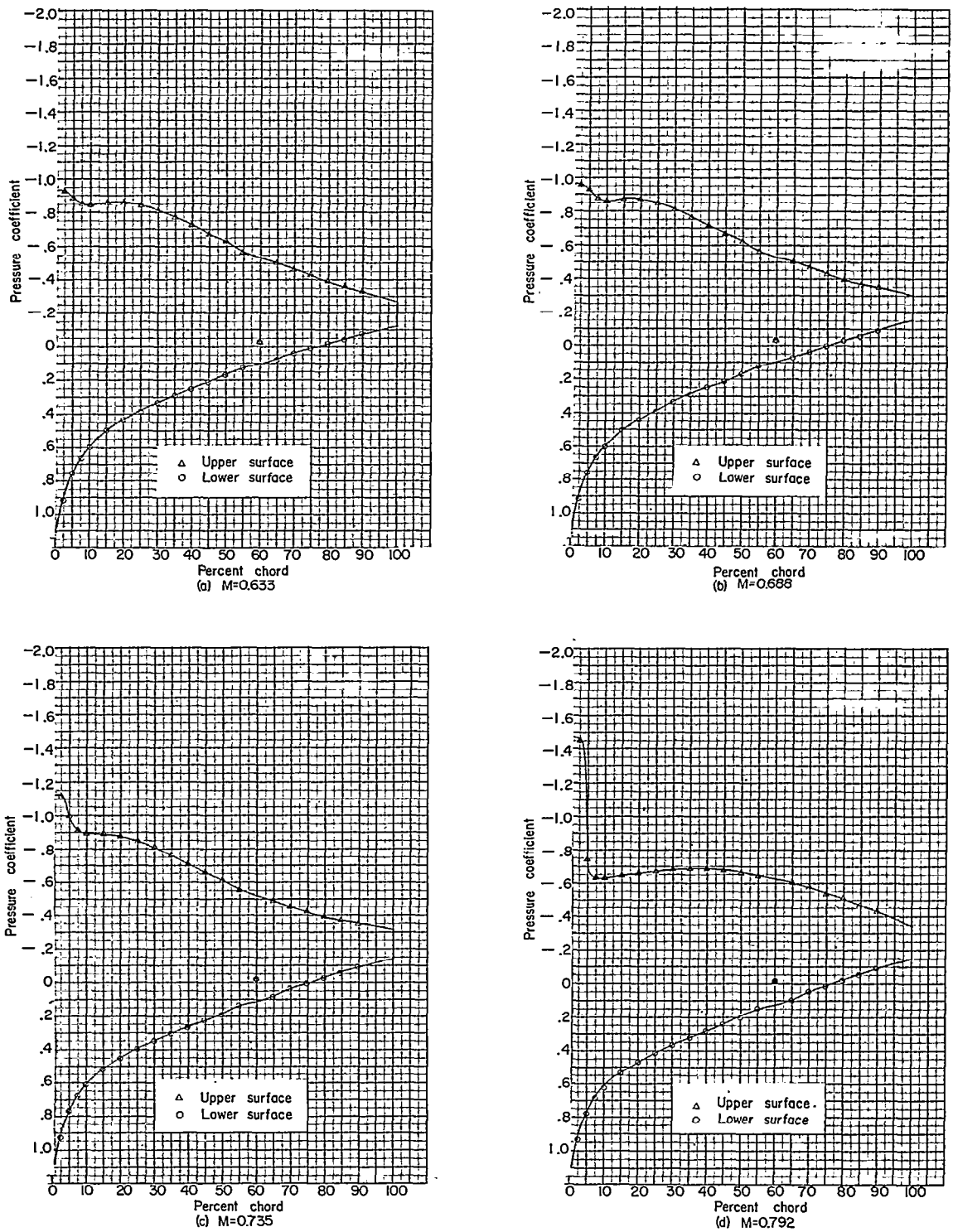
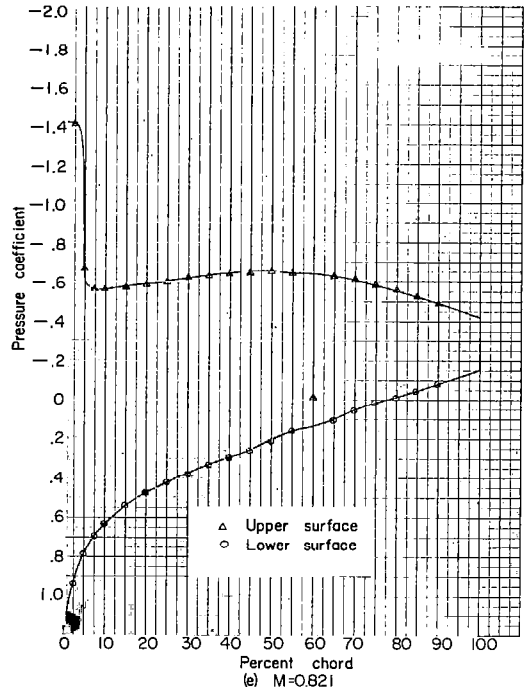
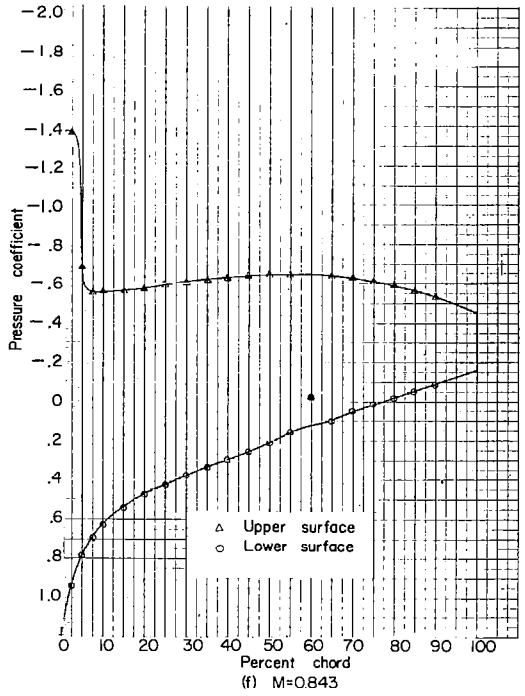
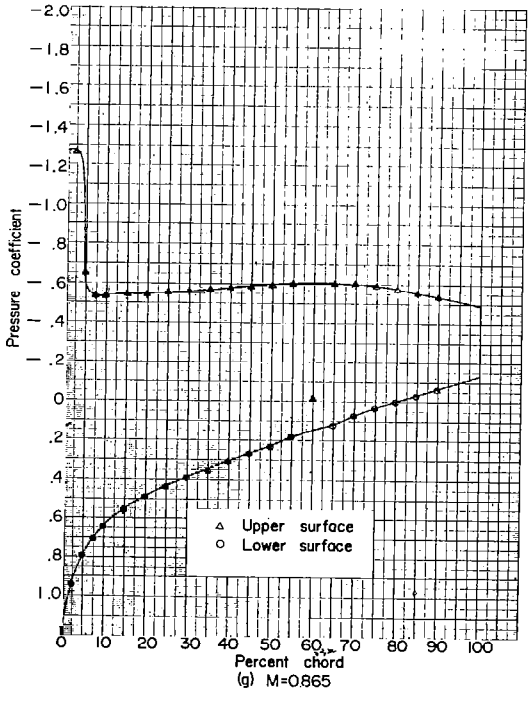


Figure 44.- Pressure distributions over NACA 16-206 airfoil section. $\alpha = 12^{\circ}$.







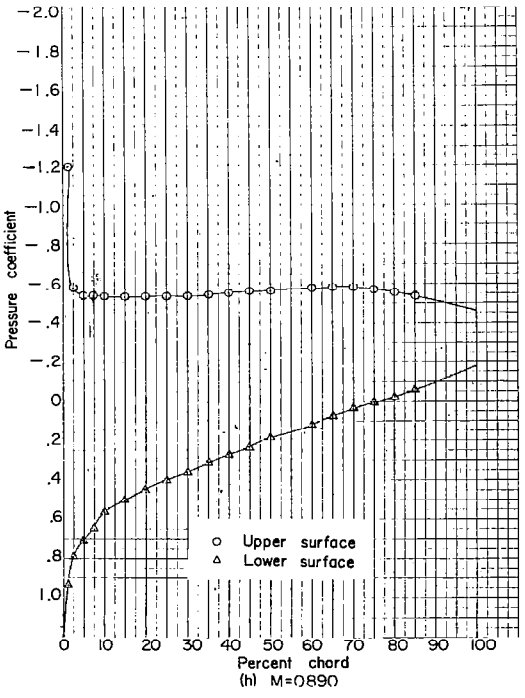
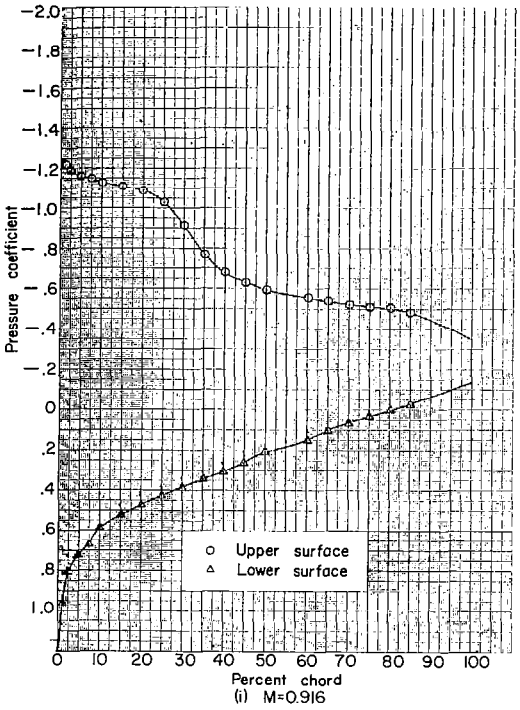
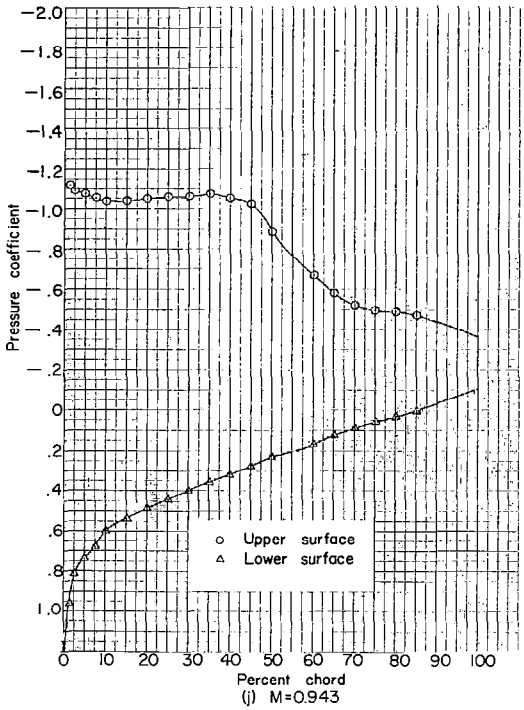
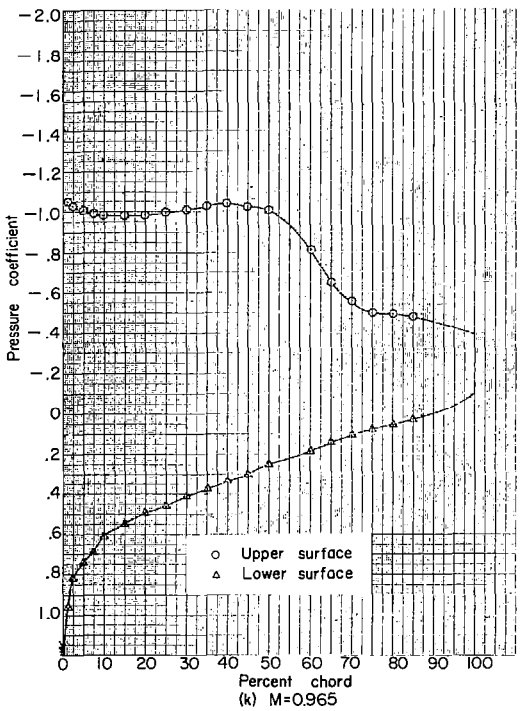


Figure 44.- Continued. NACA 16-206; $\alpha = 12^{\circ}$.







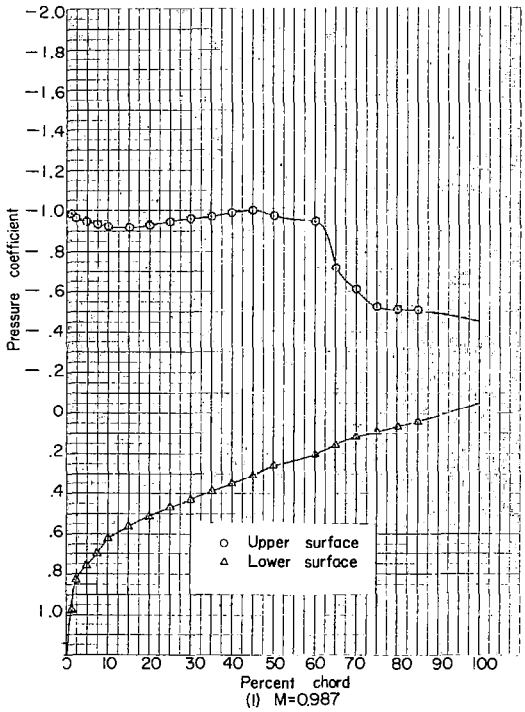
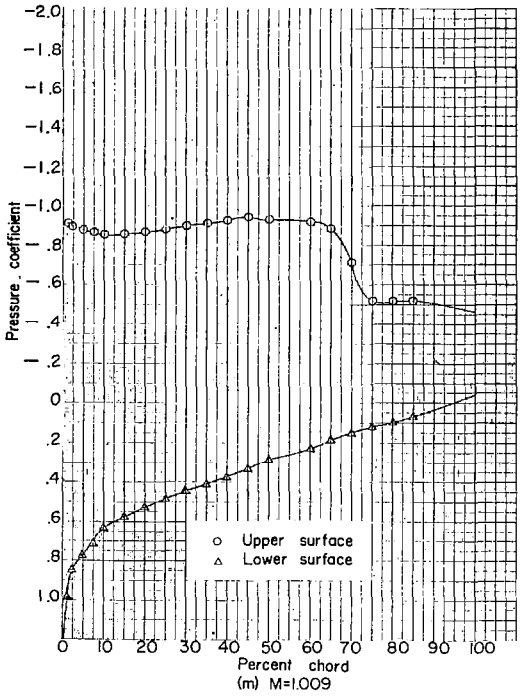
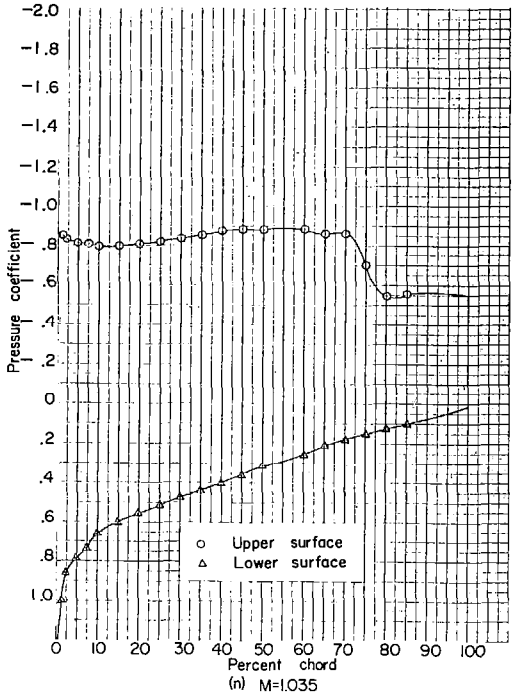
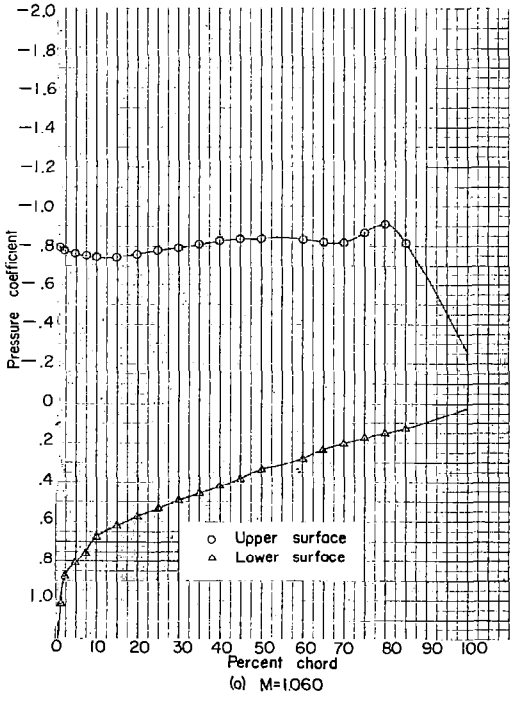


Figure 44.- Continued. NACA 16-206; $\alpha = 12^{\circ}$.







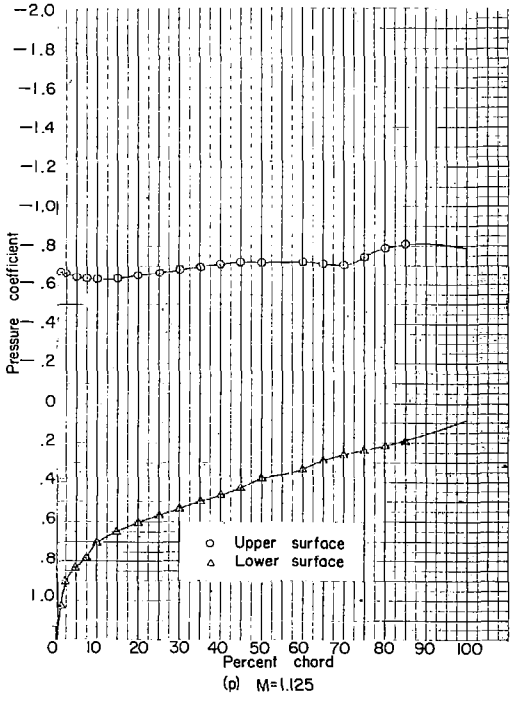


Figure 44.- Concluded. NACA 16-206; $\alpha = 12^{\circ}$.

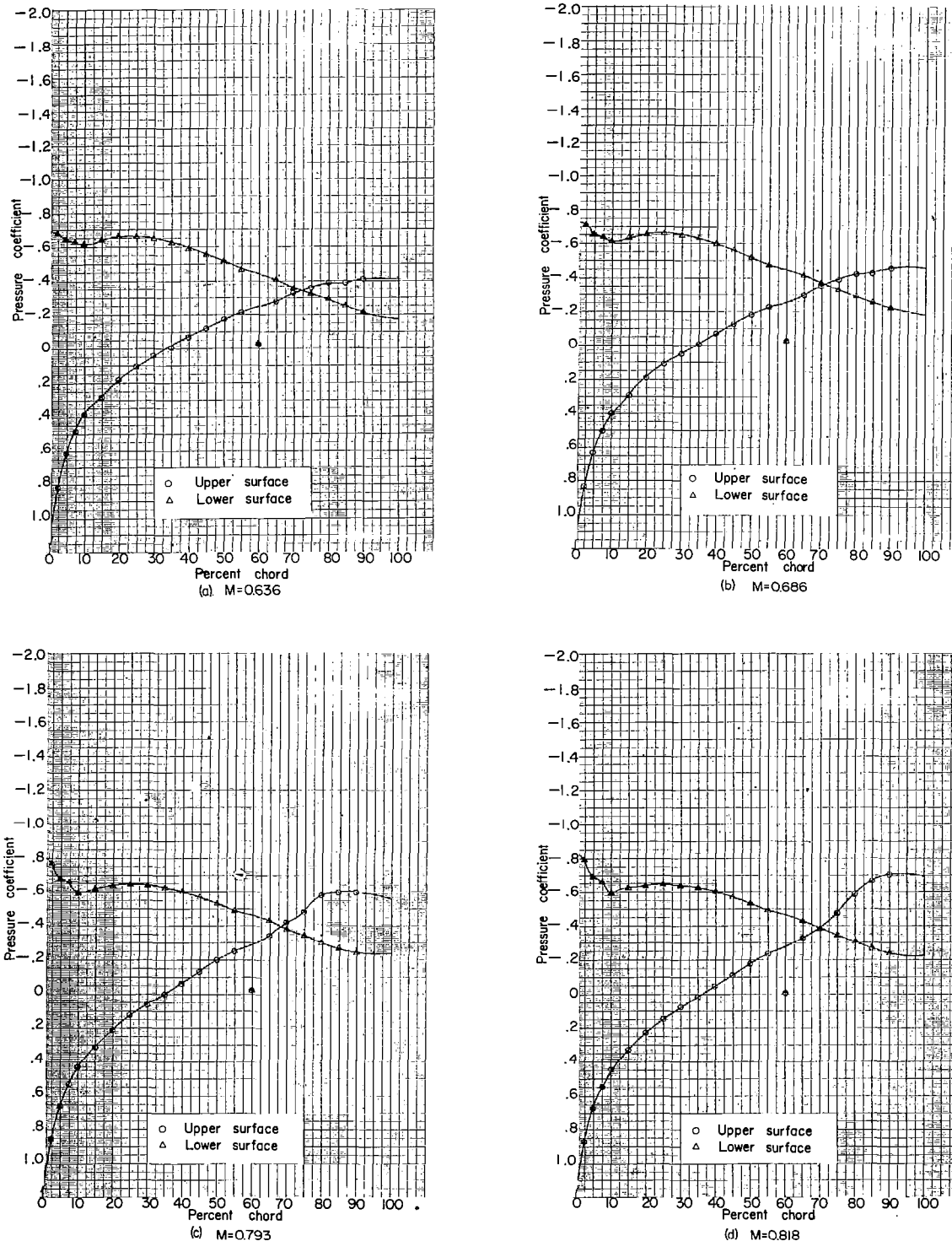
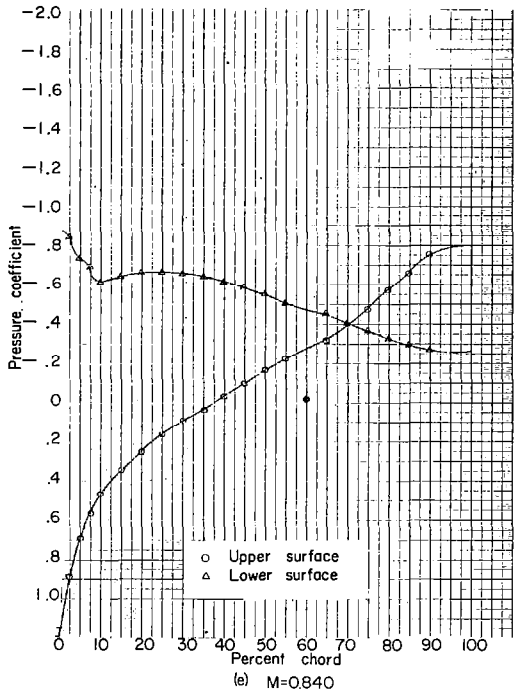
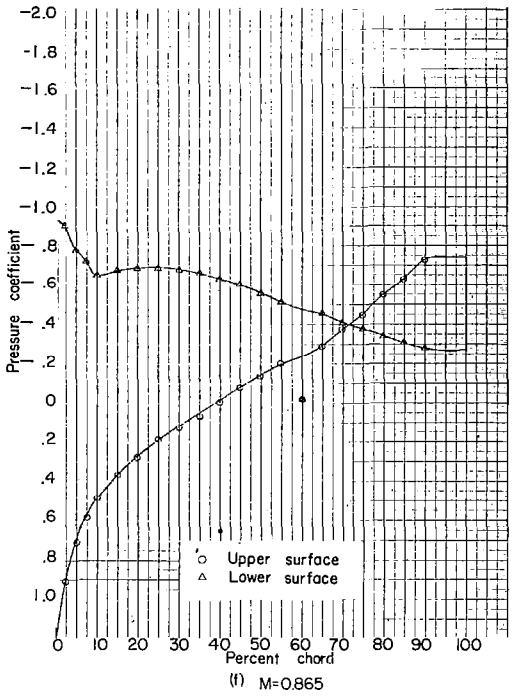


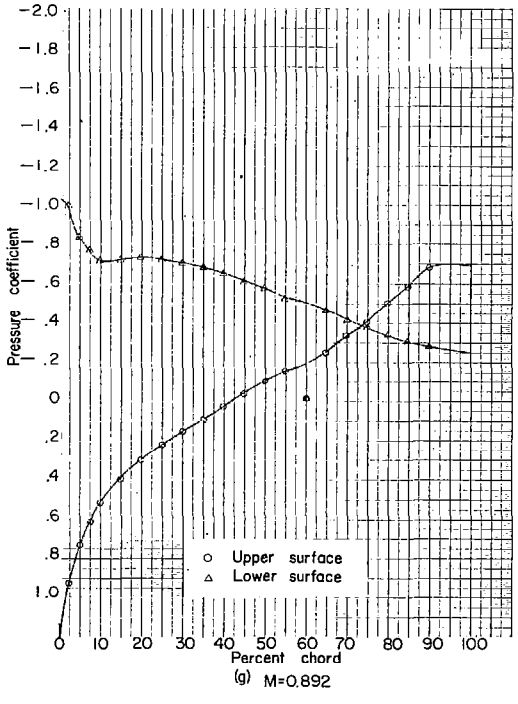
Figure 45.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = -10^{\circ}$.



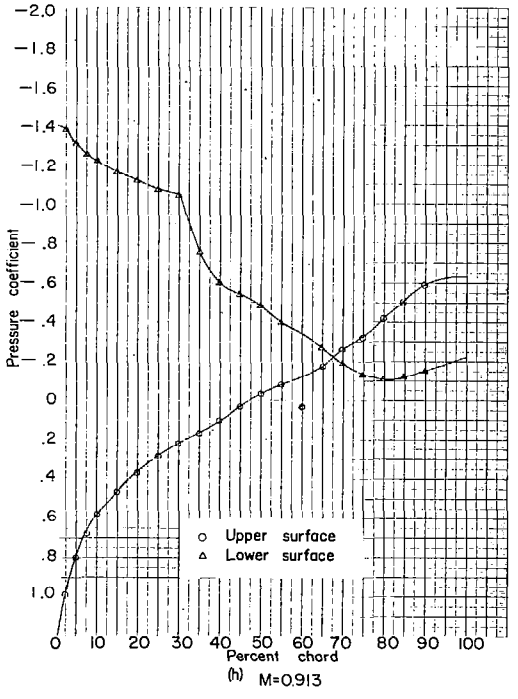
(e) M=0.840



(f) M=0.865

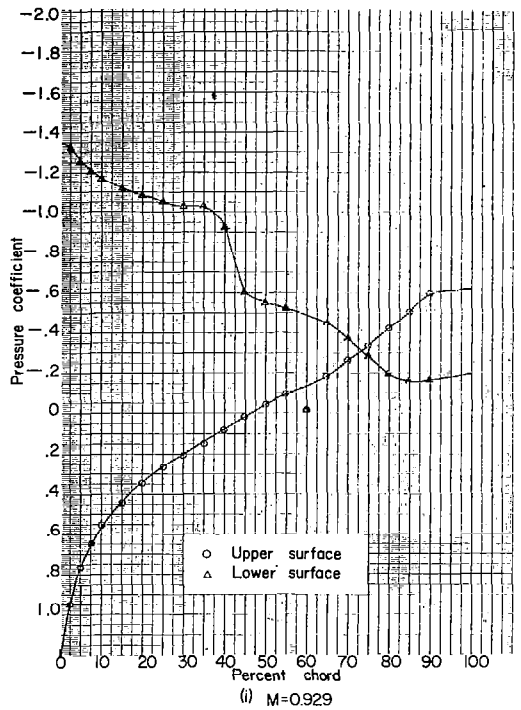


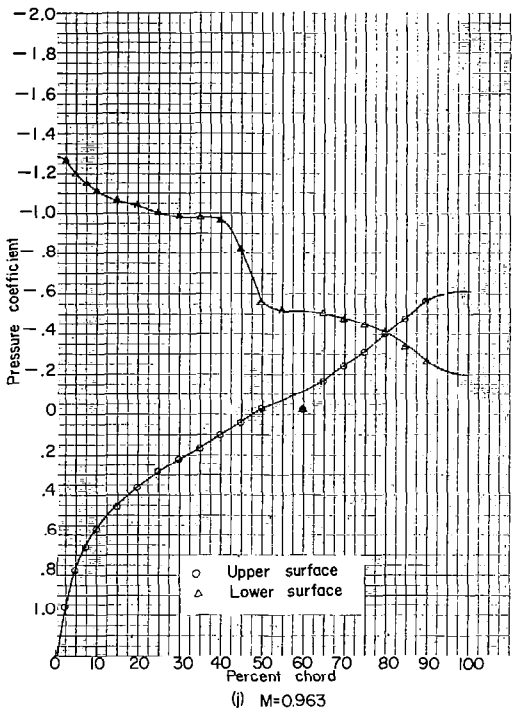
(g) M=0.892

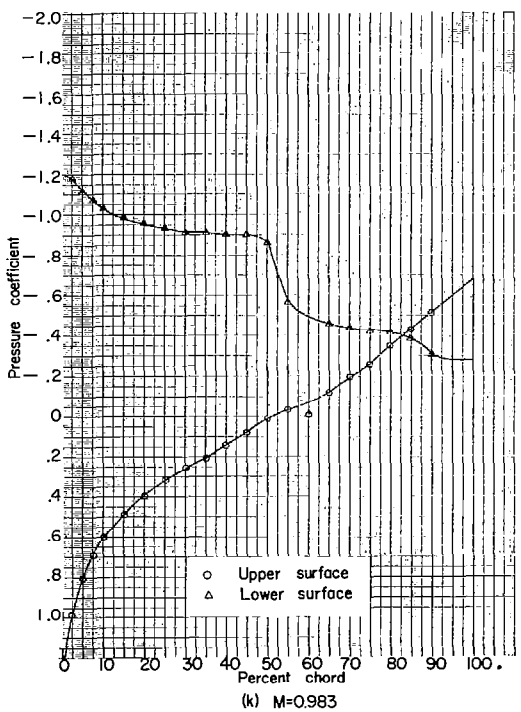


(h) M=0.913

Figure 45.- Continued. NACA 16-506; $\alpha = -10^{\circ}$.







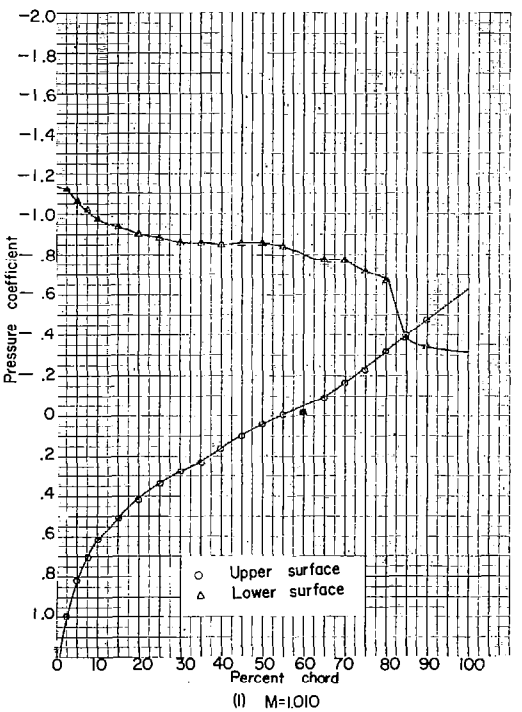
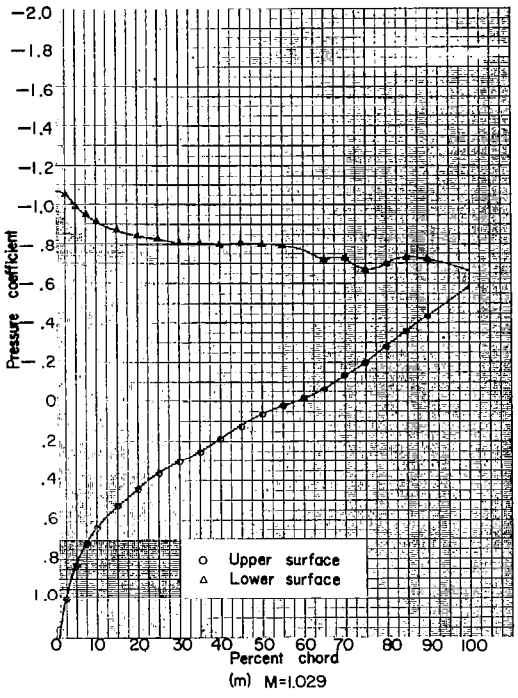
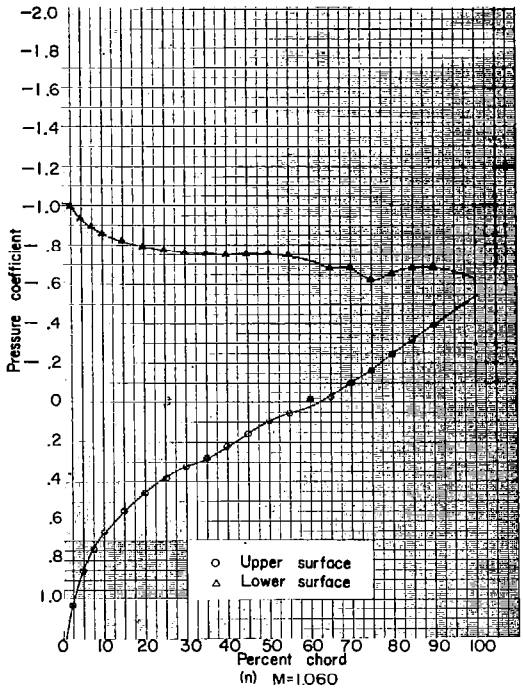
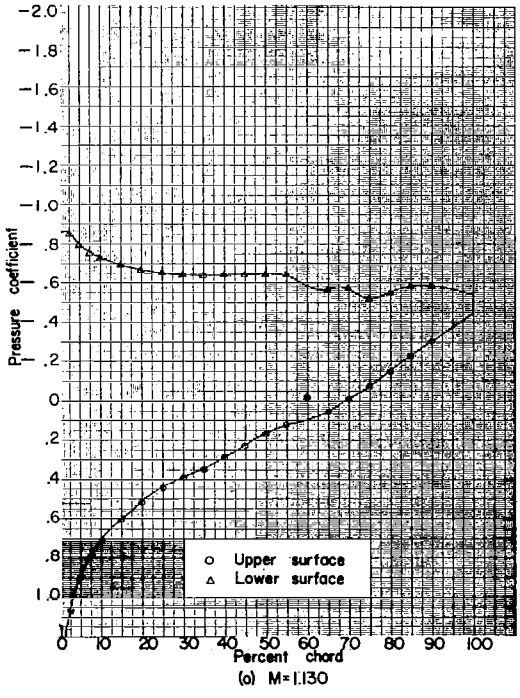


Figure 45.- Continued. NACA 16-506; $\alpha = -10^{\circ}$.







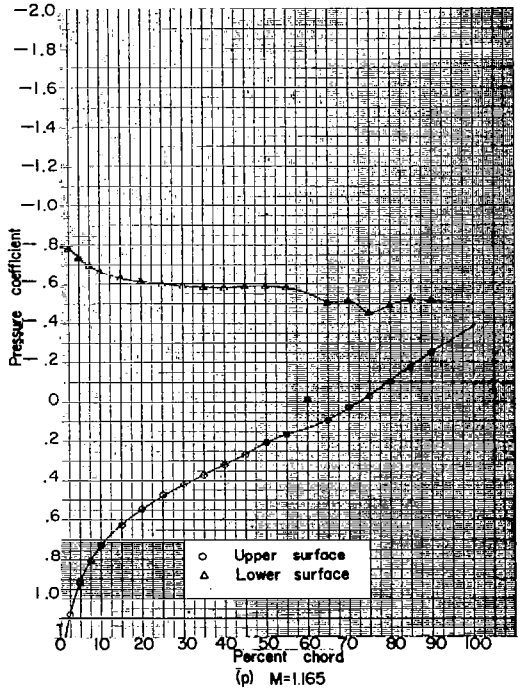


Figure 45.- Concluded. NACA 16-506; $\alpha = -10^{\circ}$.

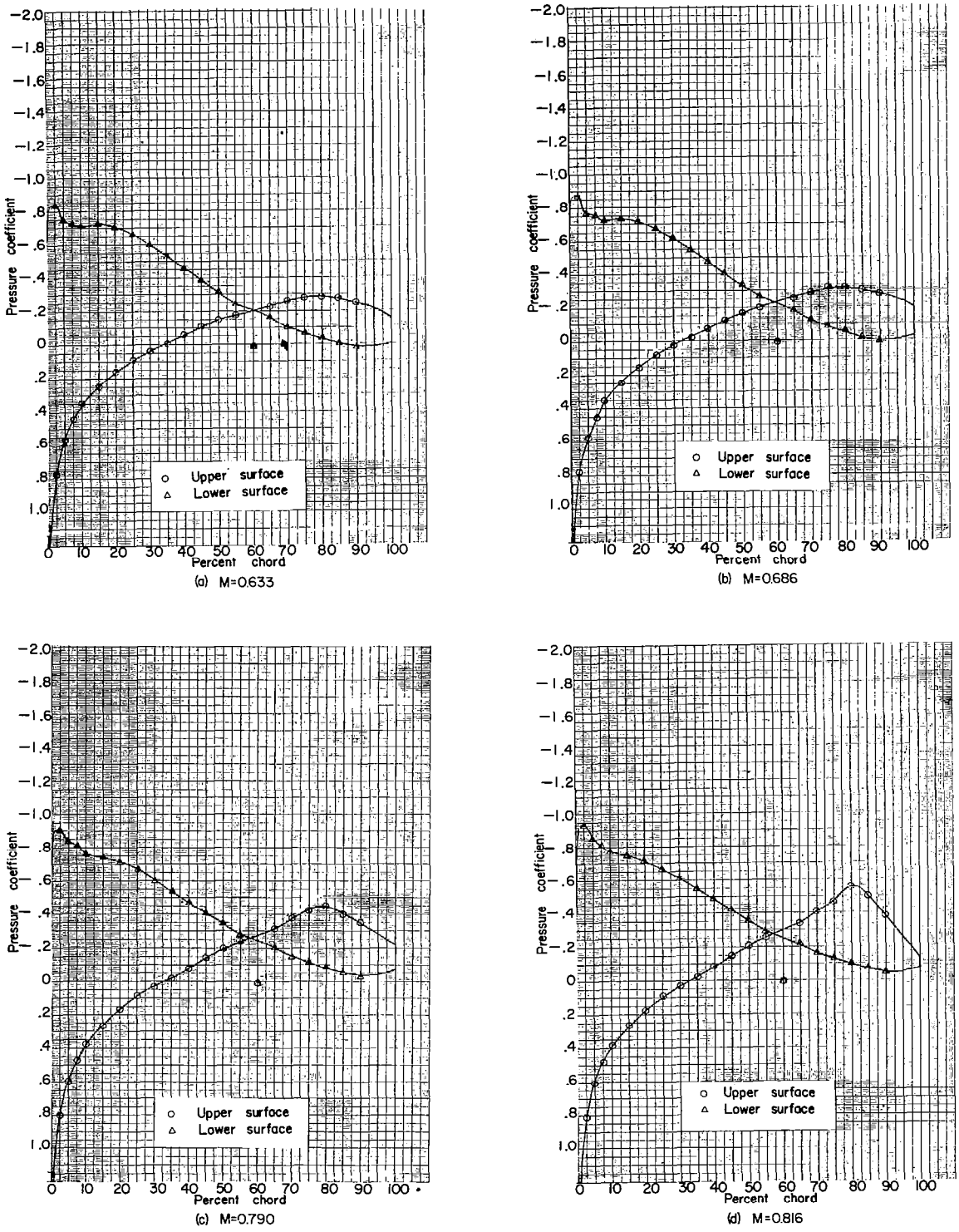
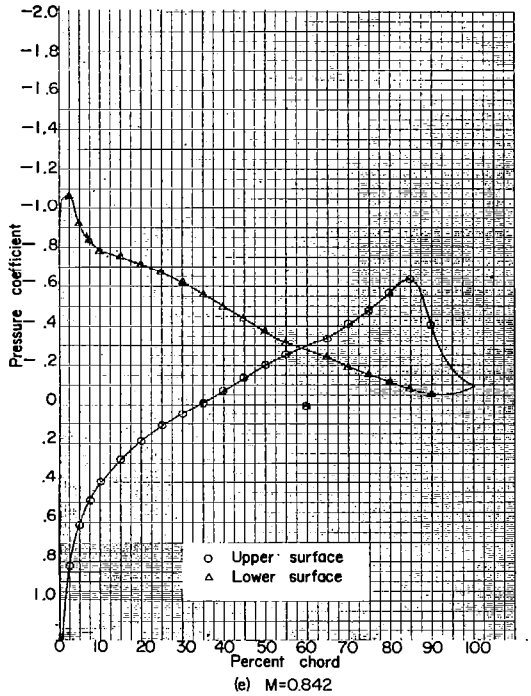
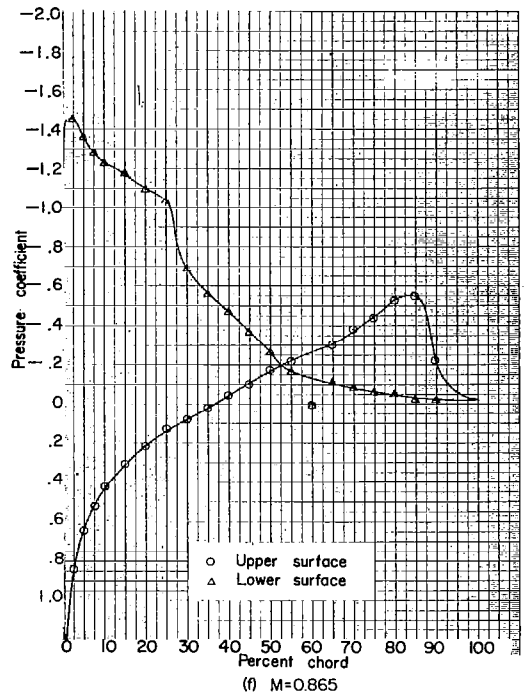


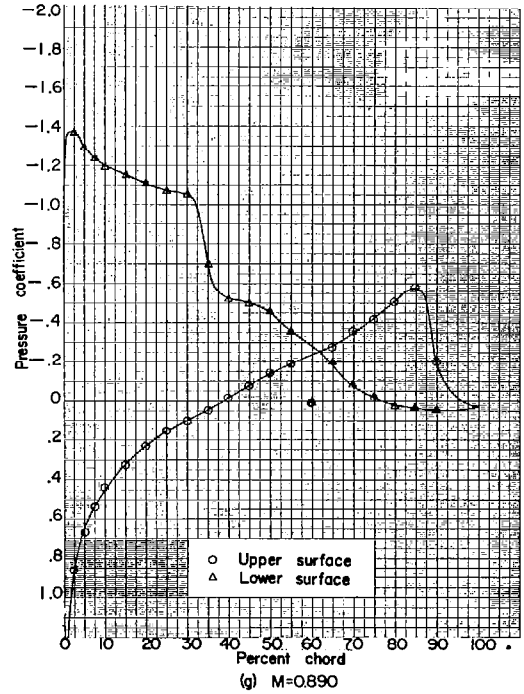
Figure 46.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = -8^{\circ}$.



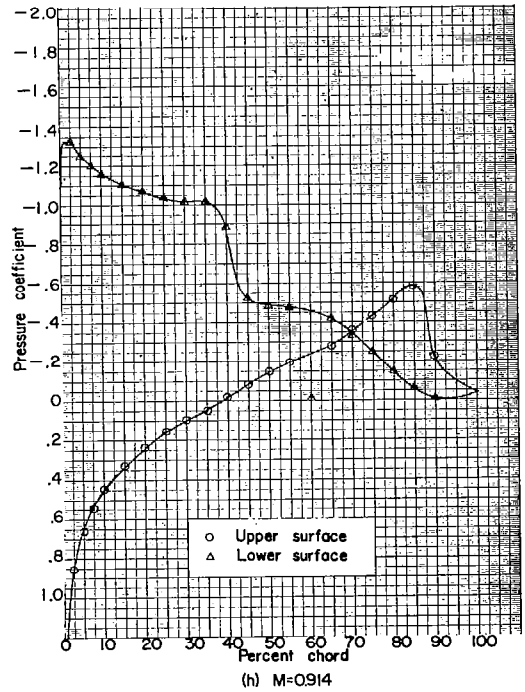
(e) M=0.842



(f) M=0.865

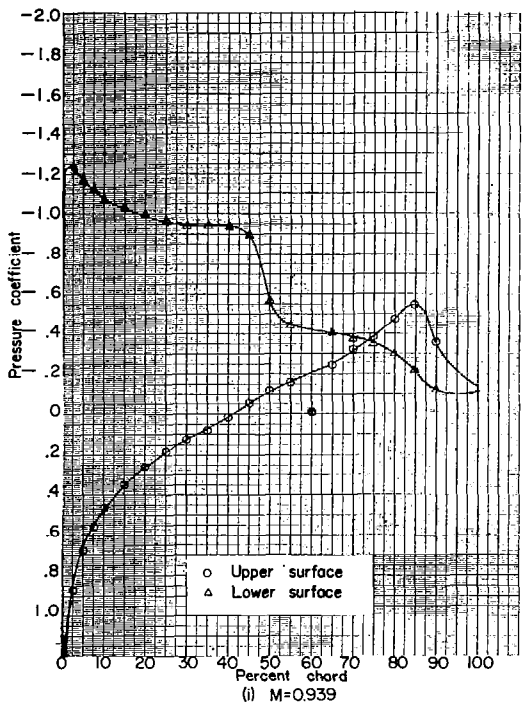


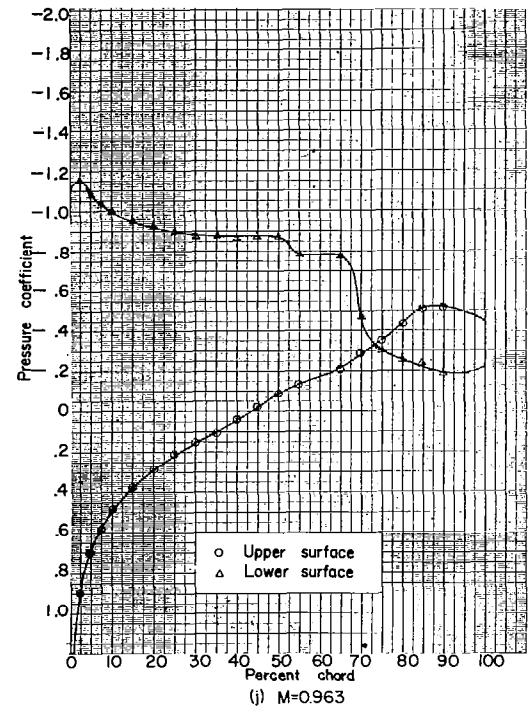
(g) M=0.890

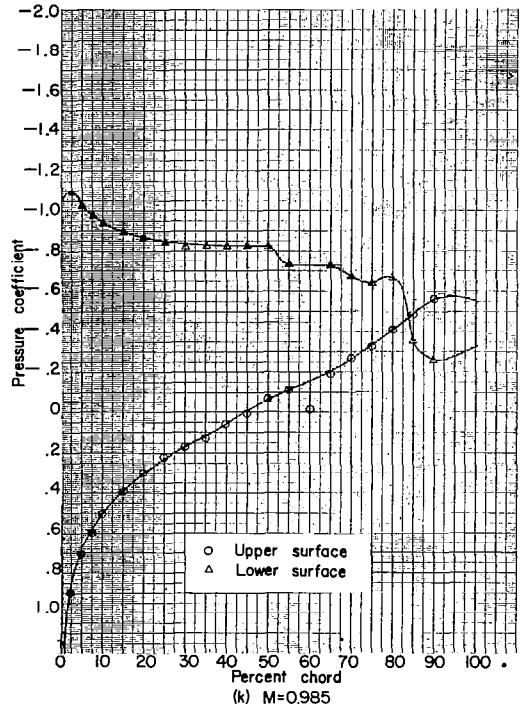


(h) M=0.914

Figure 46.- Continued. NACA 16-506; $\alpha = -8^{\circ}$.







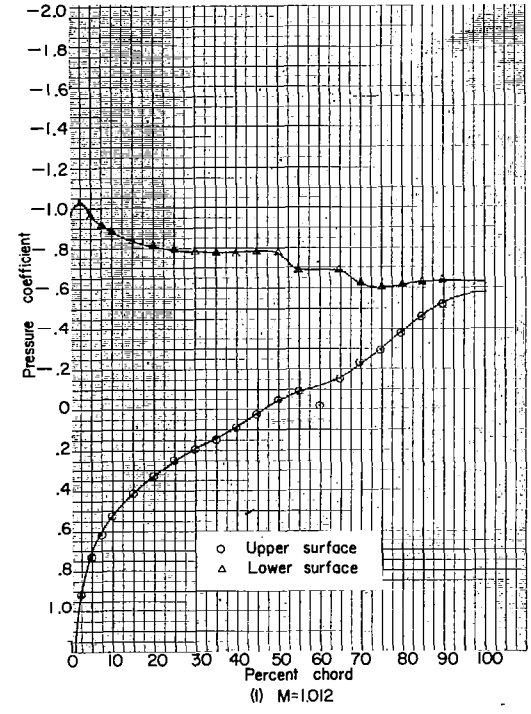
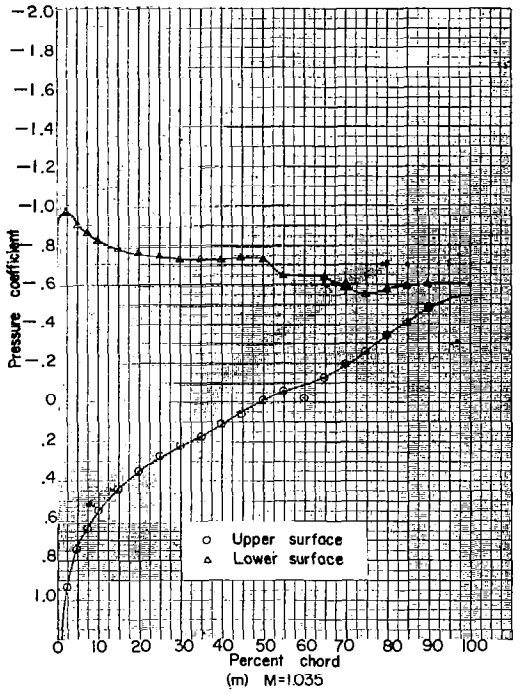
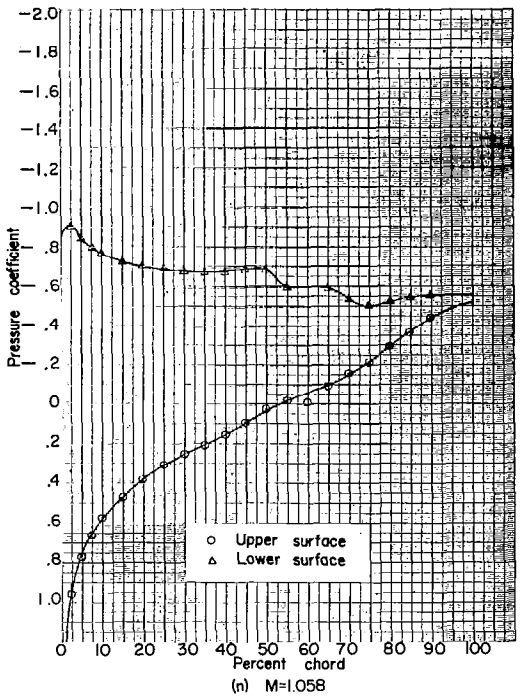


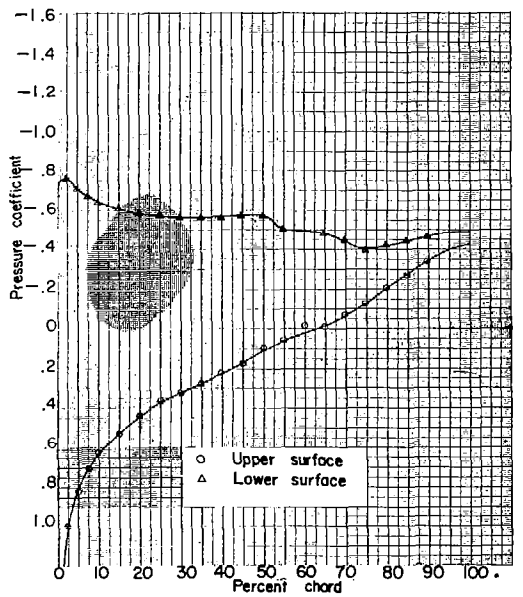
Figure 46.- Continued. NACA 16-506; $\alpha = -8^{\circ}$.



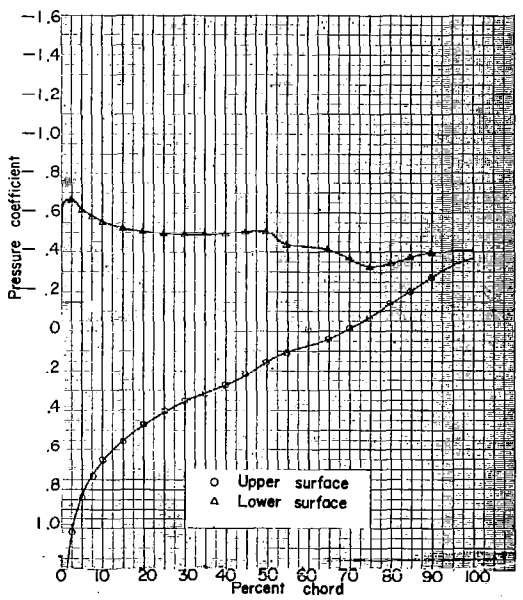
(m) M=1.035



(n) M=1.058



(o) M=1.132



(p) M=1.172

Figure 46.- Concluded. NACA 16-506; $\alpha = -8^{\circ}$.

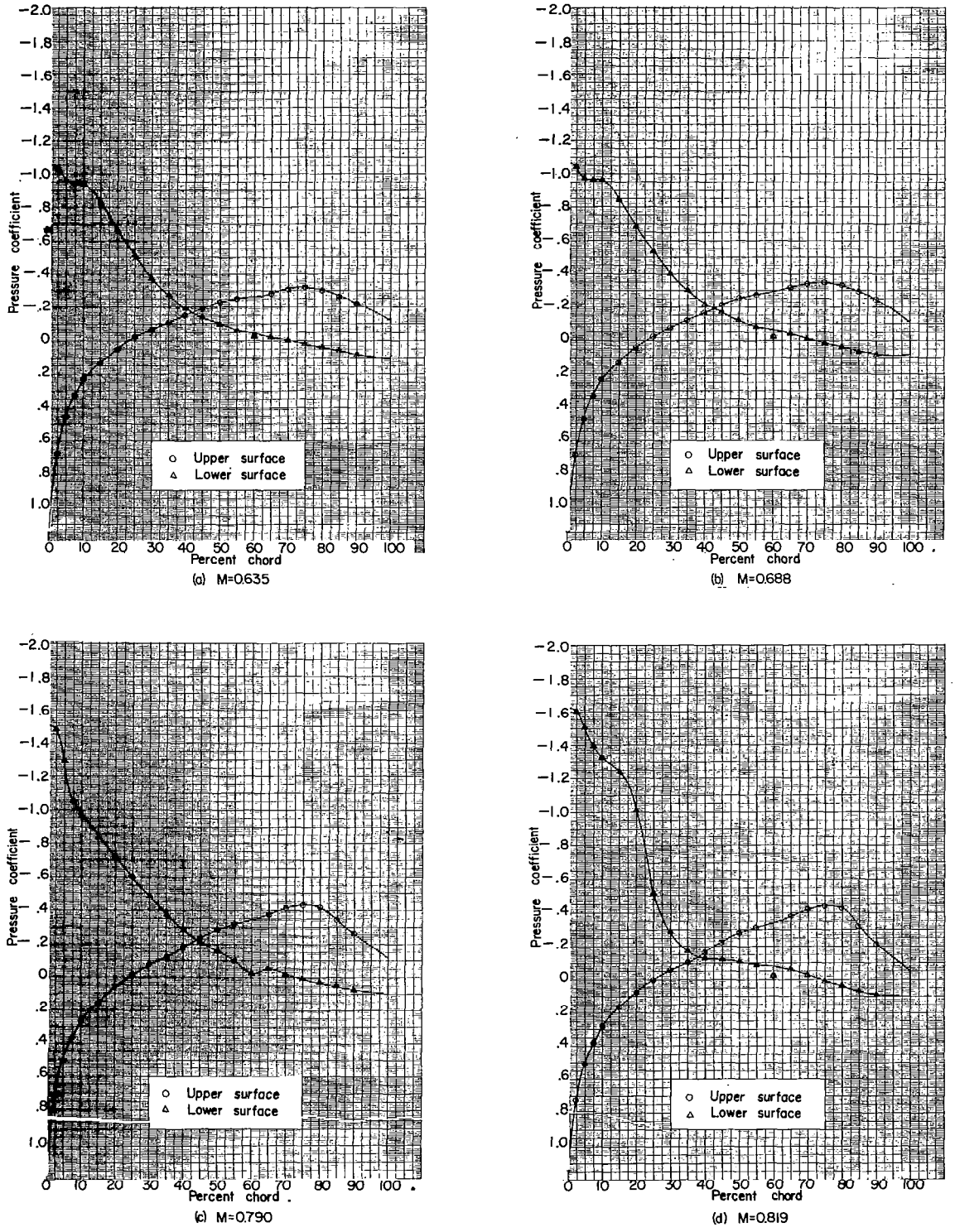
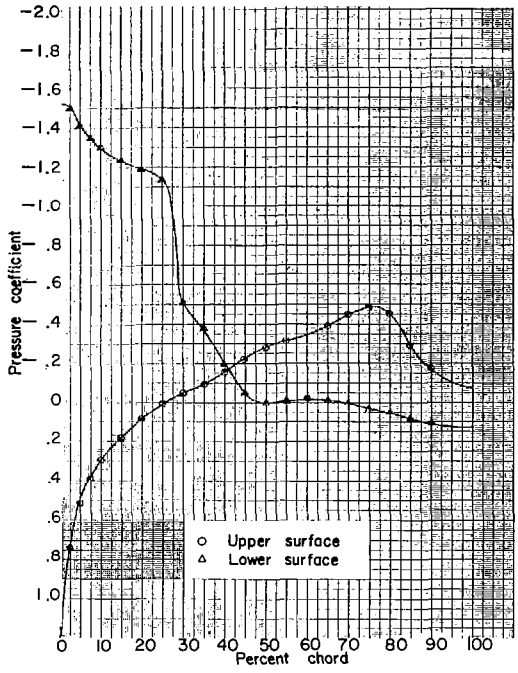
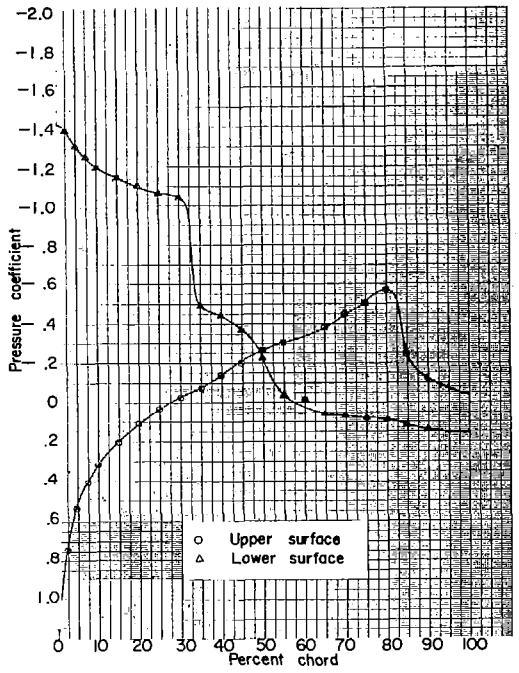


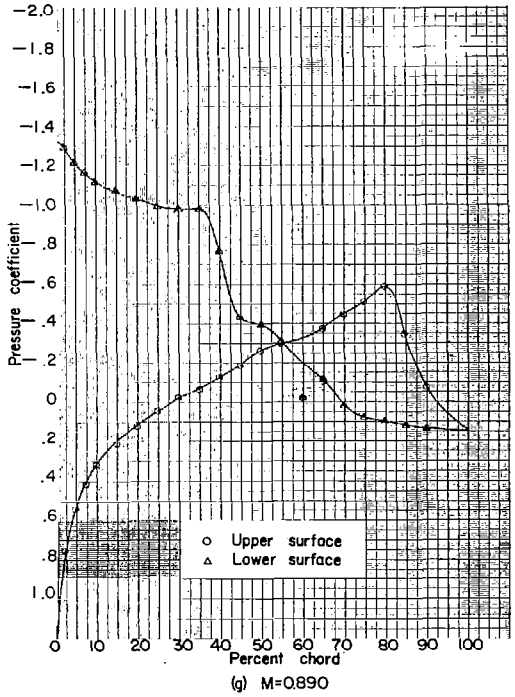
Figure 47.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = -6^{\circ}$.



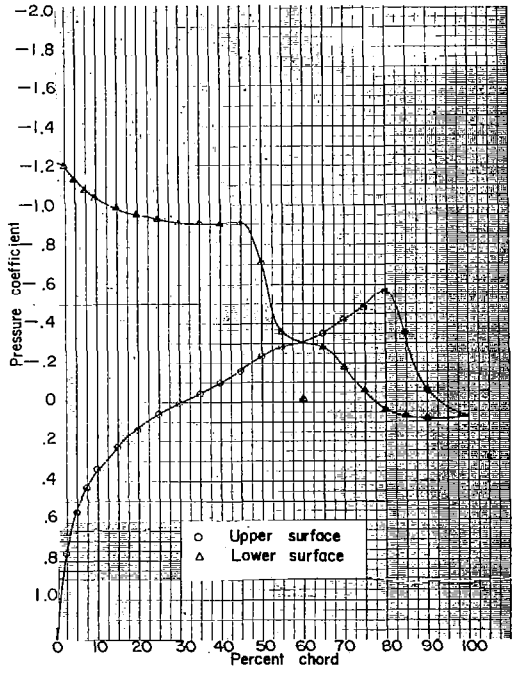
(e) M=0.843



(f) M=0.863

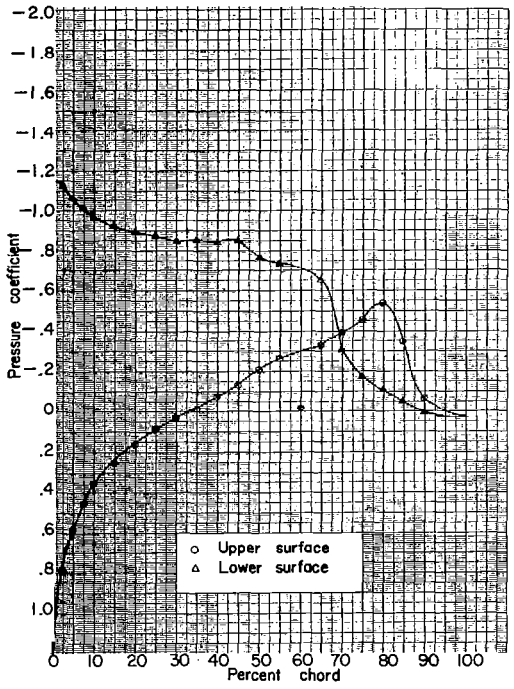


(g) M=0.890

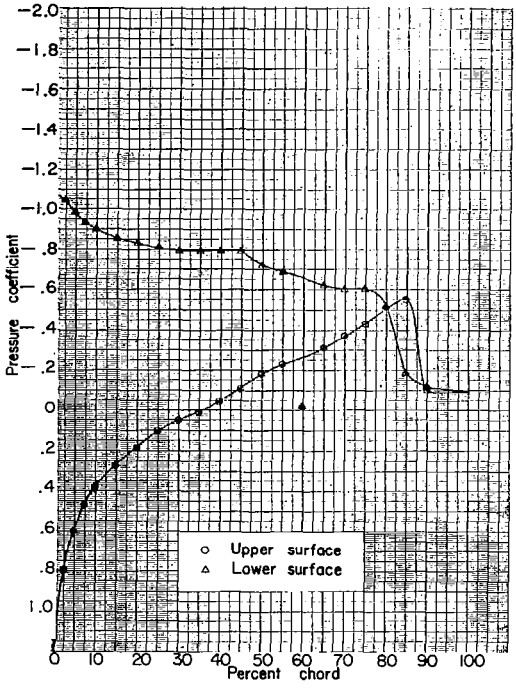


(h) M=0.913

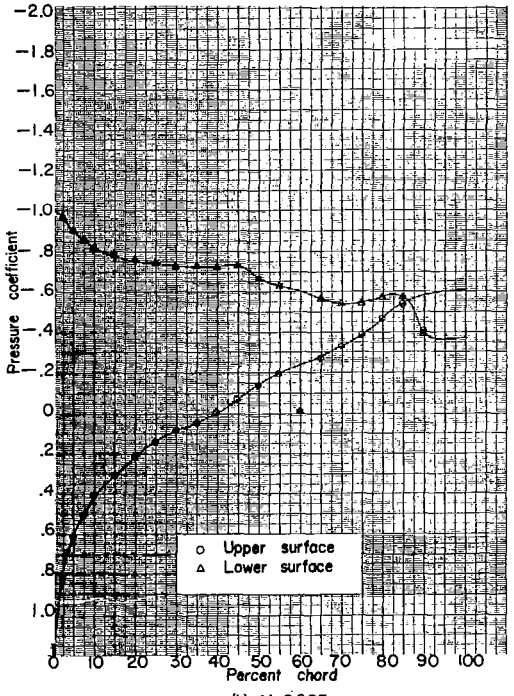
Figure 47.- Continued. NACA 16-506; $\alpha = -6^{\circ}$.



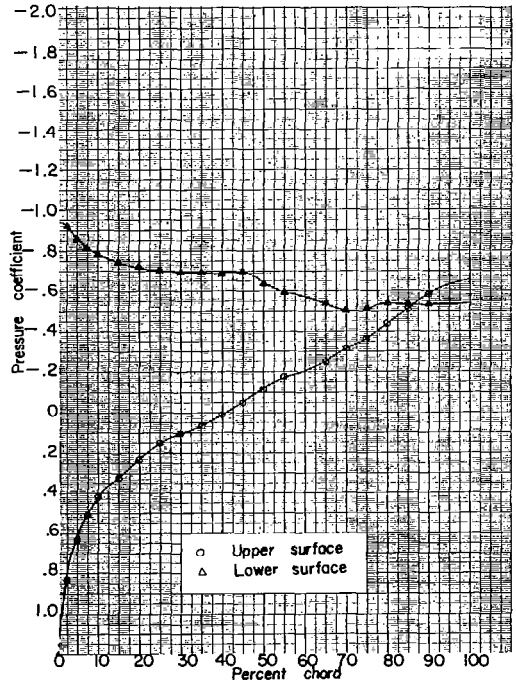
(i) M=0.941



(j) M=0.963

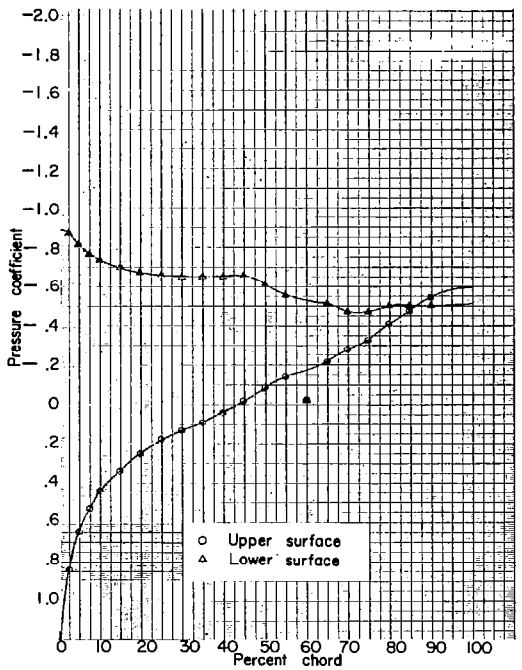


(k) M=0.985

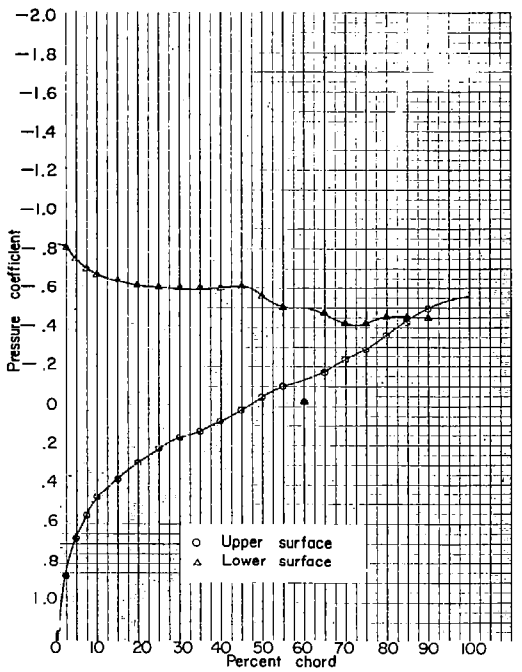


(l) M=1.010

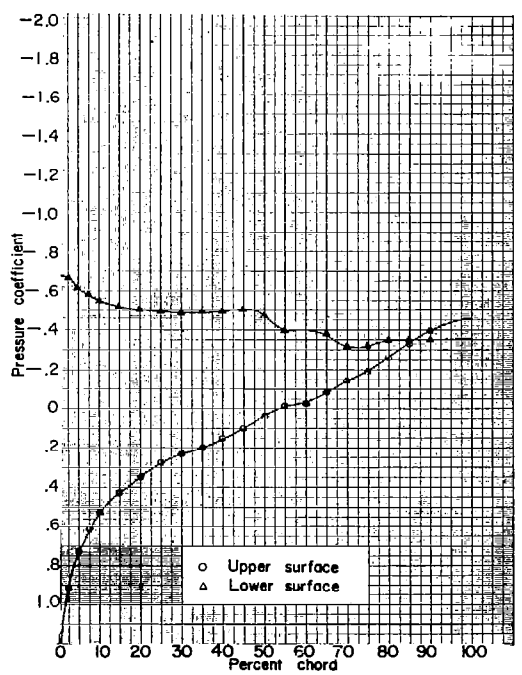
Figure 47.- Continued. NACA 16-506; $\alpha = -6^{\circ}$.



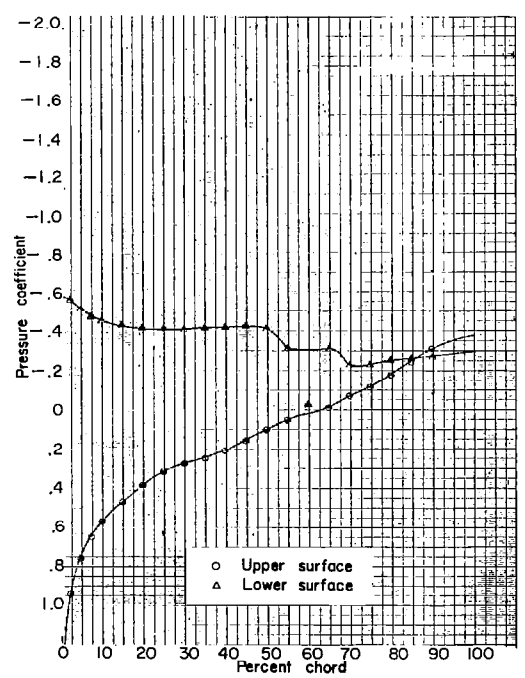
(m) M=1.029



(n) M=1.060



(o) M=1.130



(p) M=1.208

Figure 47.- Concluded. NACA 16-506; $\alpha = -6^{\circ}$.

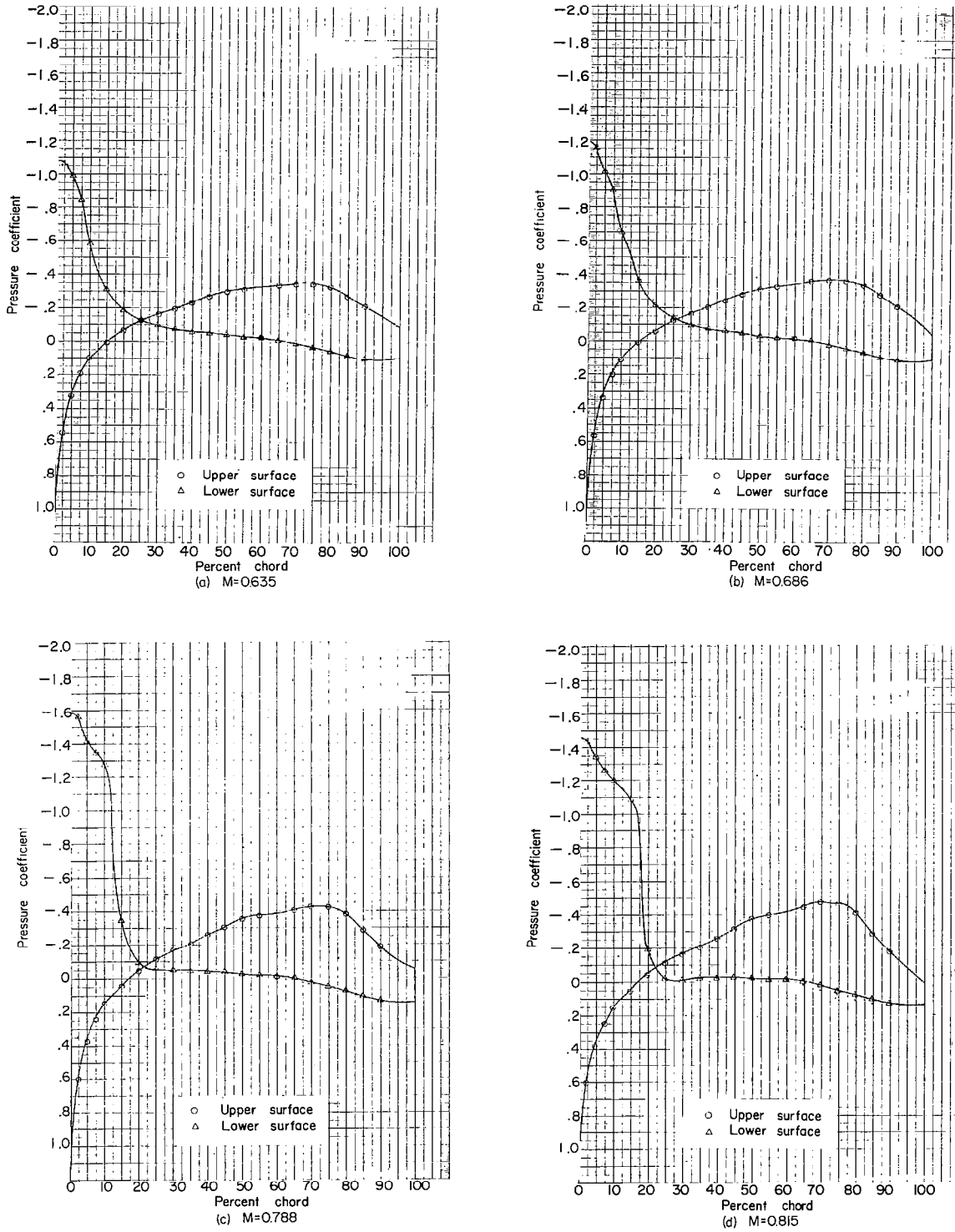
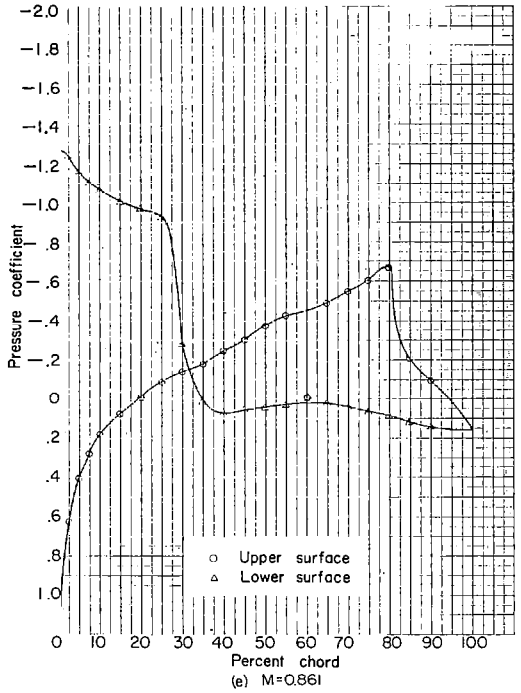
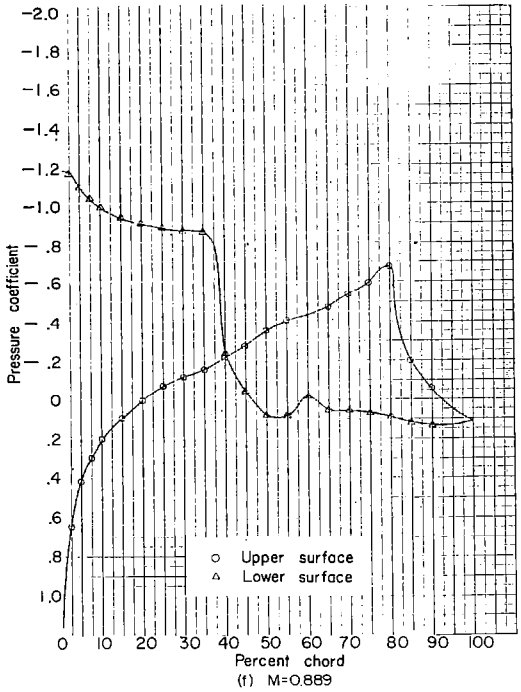
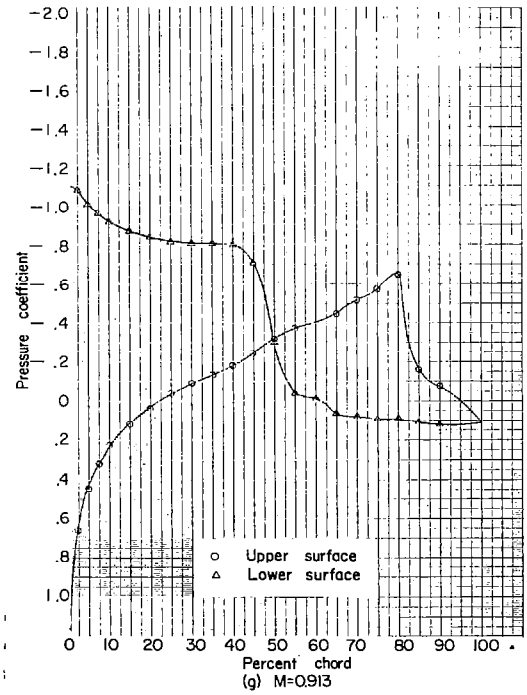


Figure 48.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = -4^{\circ}$.







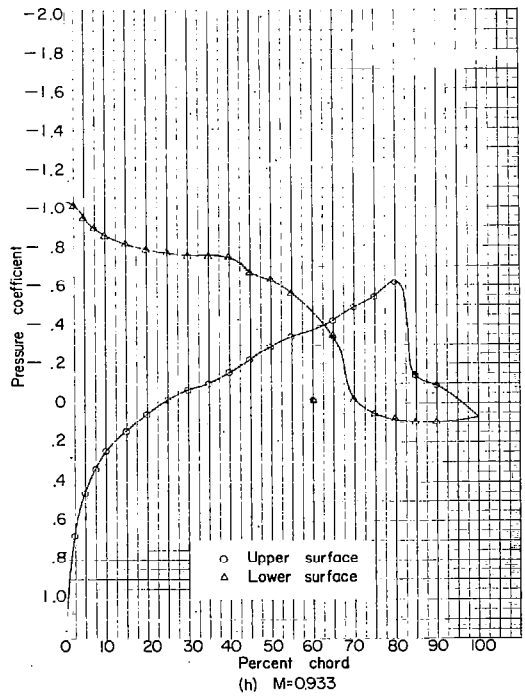
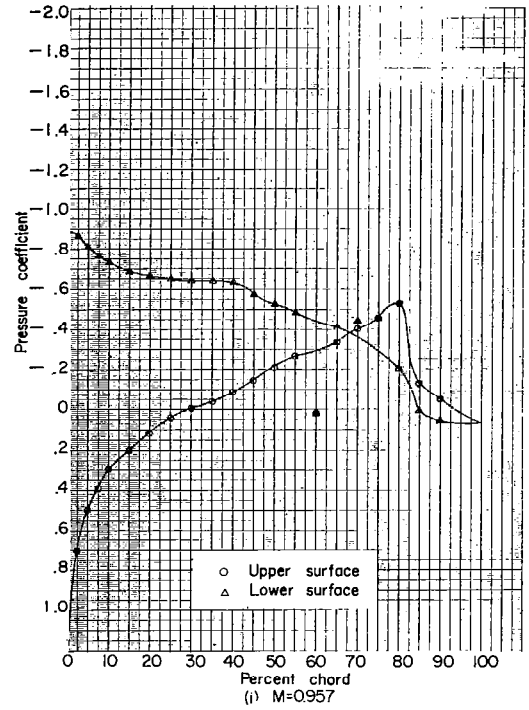
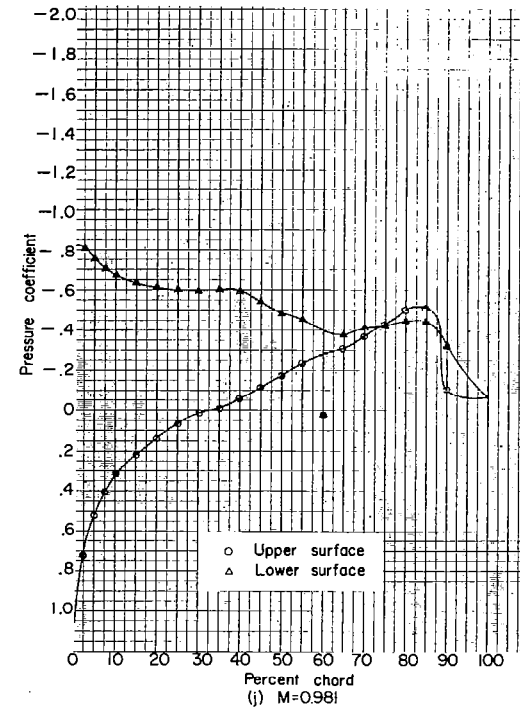
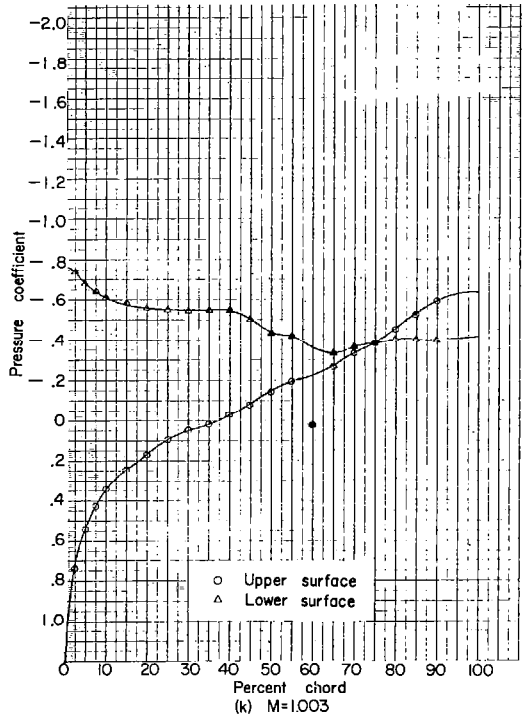


Figure 48.- Continued. NACA 16-506; $\alpha = -4^{\circ}$.







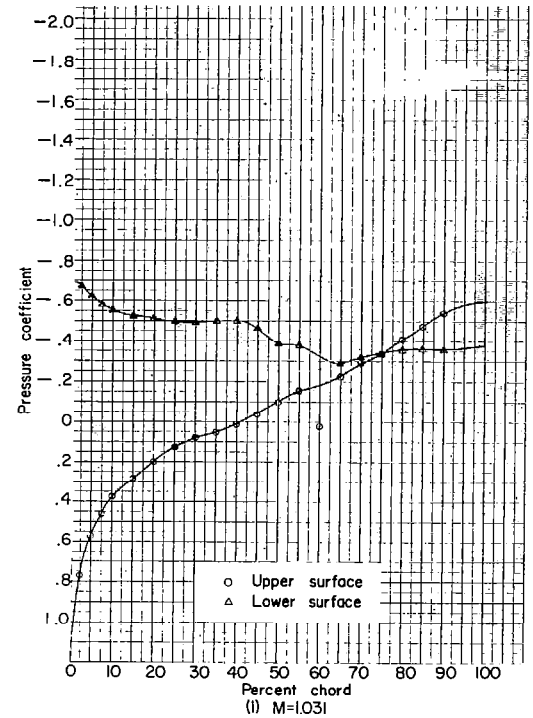
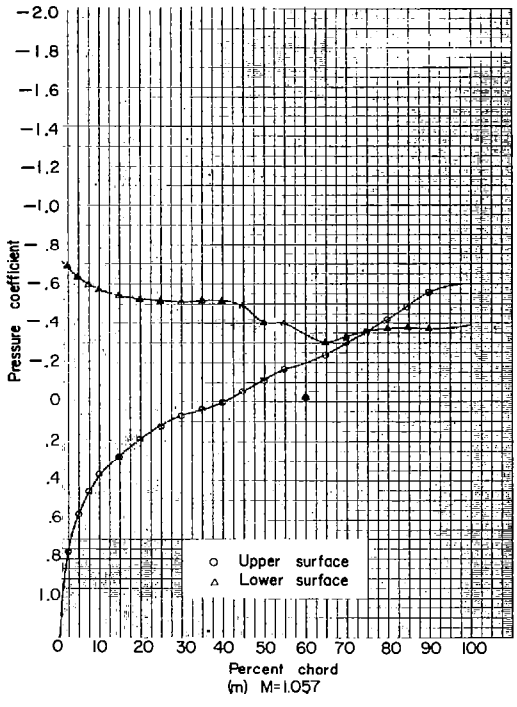
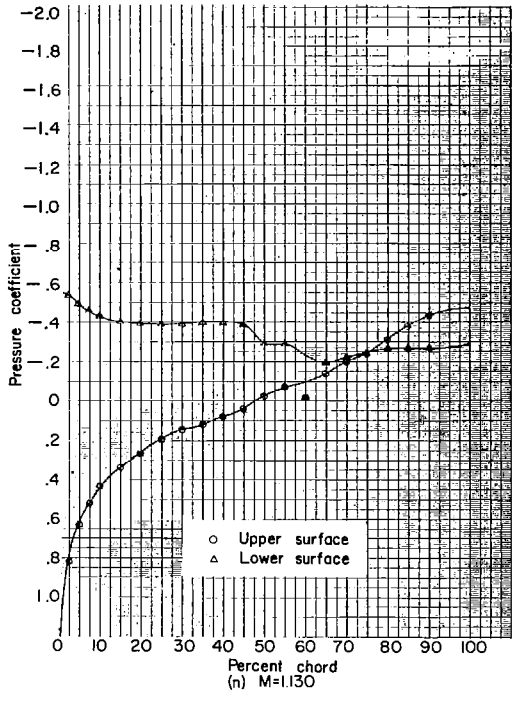
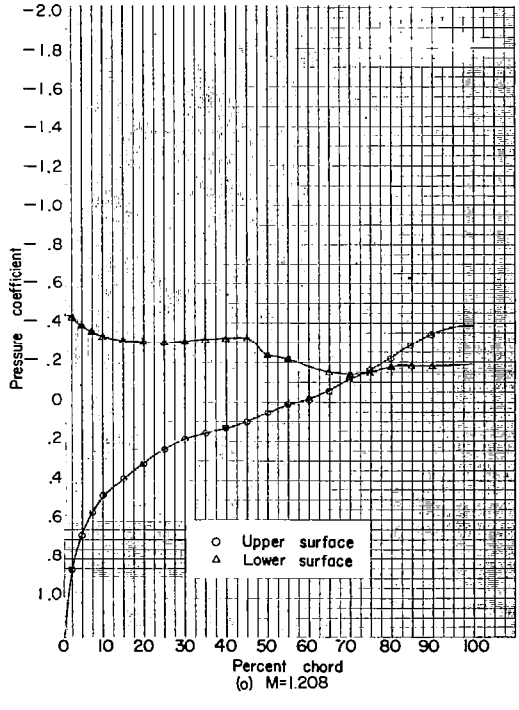


Figure 48.- Continued. NACA 16-506; $\alpha = -4^{\circ}$.







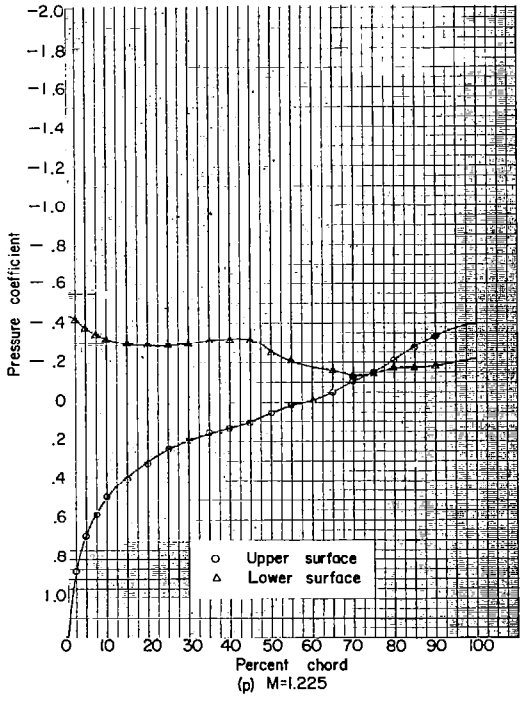


Figure 48.- Concluded. NACA 16-506; $\alpha = -4^{\circ}$.

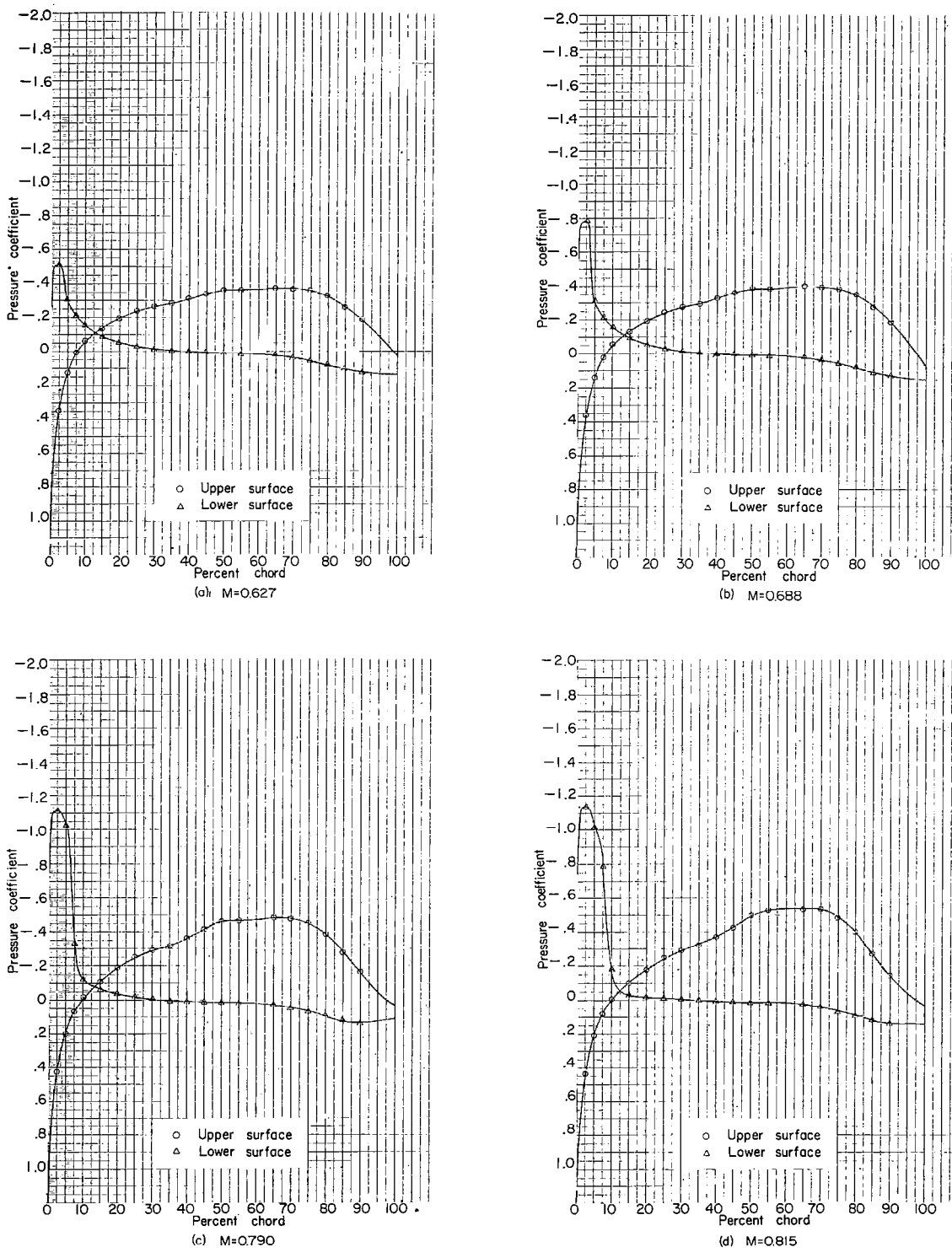
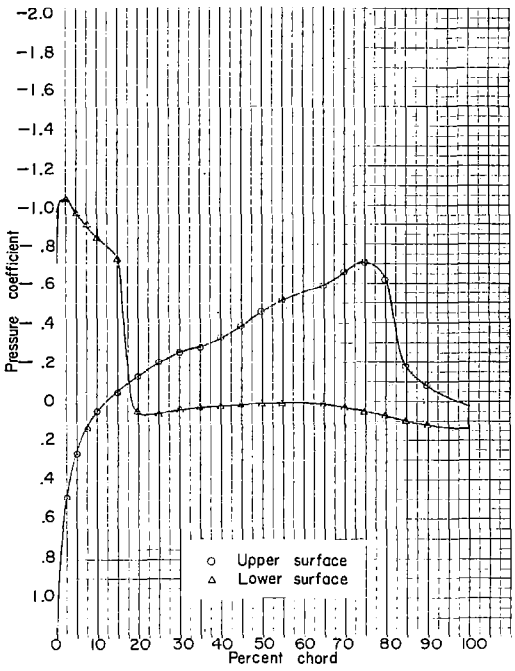
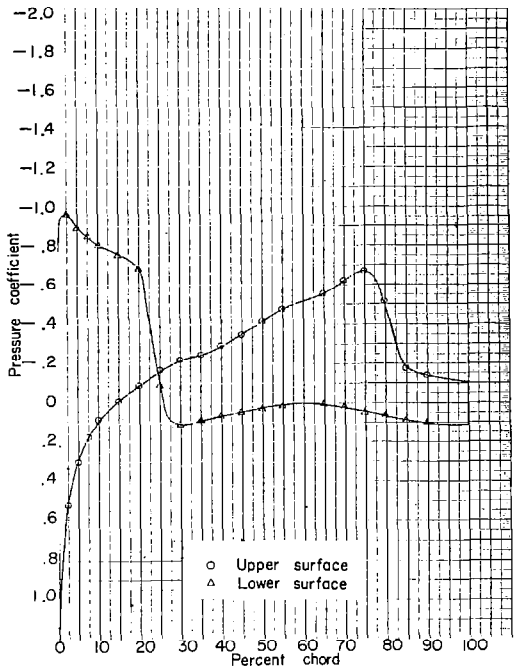


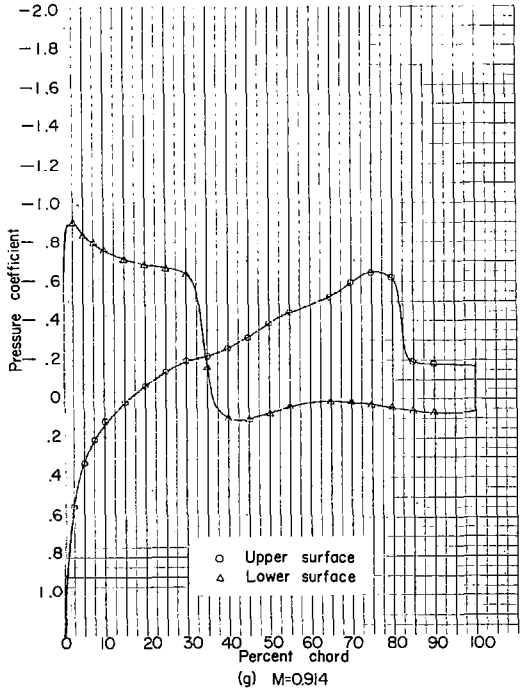
Figure 49.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = -2^{\circ}$.



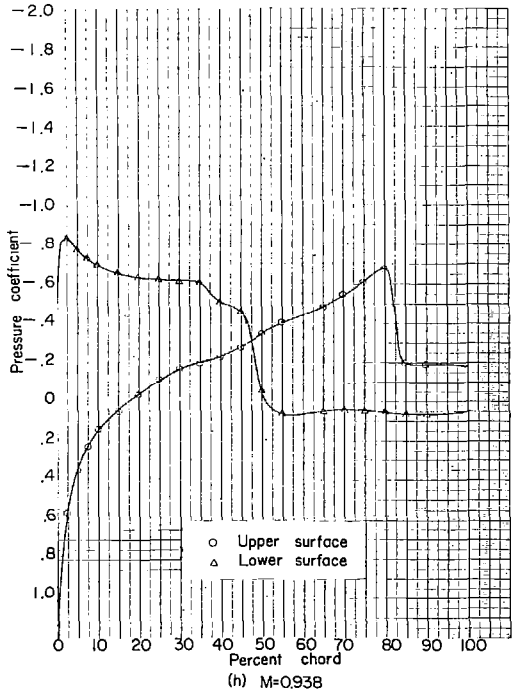
(e) M=0863



(f) M=0889

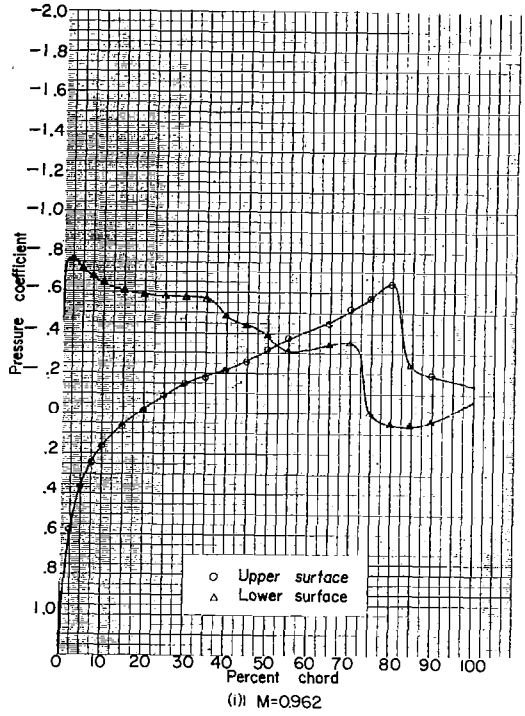


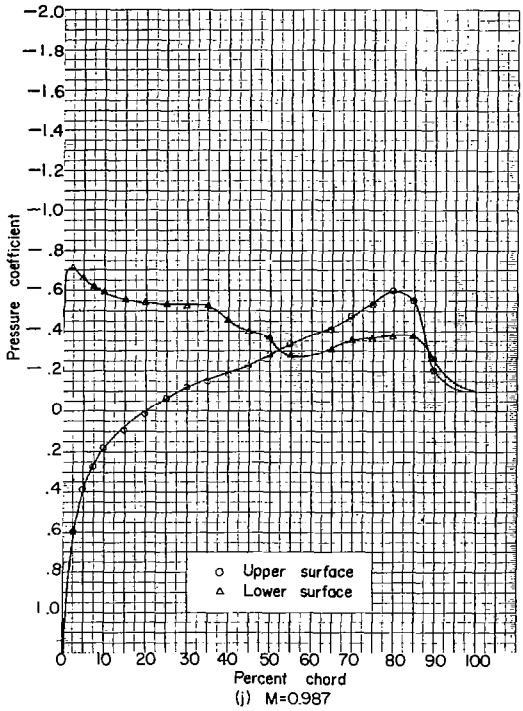
(g) M=0914

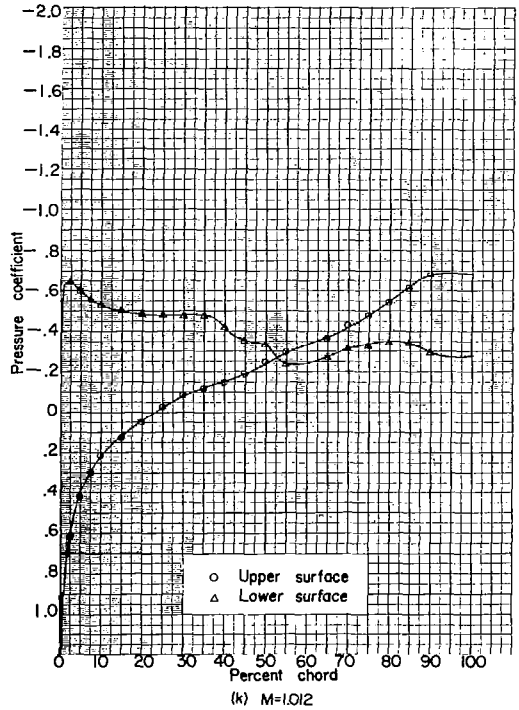


(h) M=0938

Figure 49.- Continued. NACA 16-506; $\alpha = -2^{\circ}$.







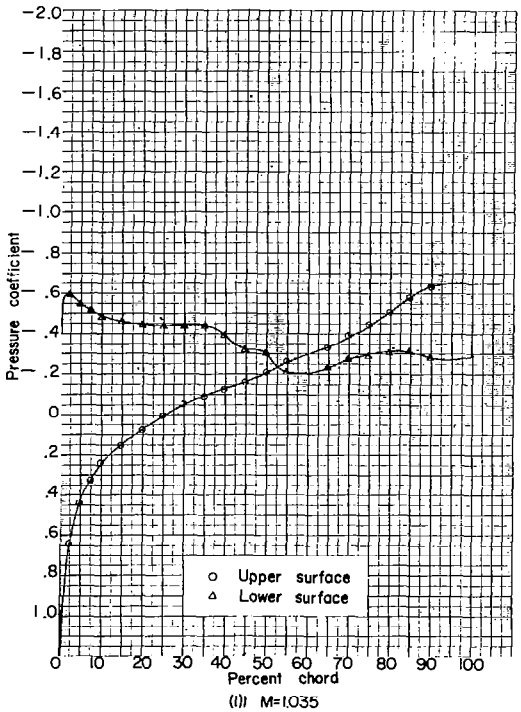
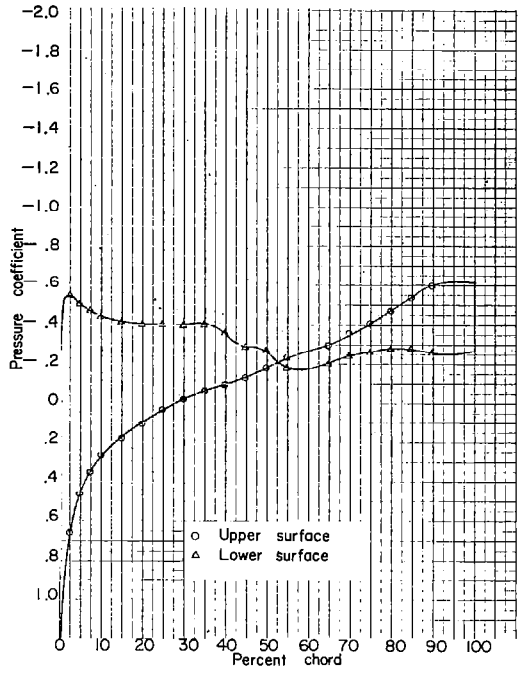
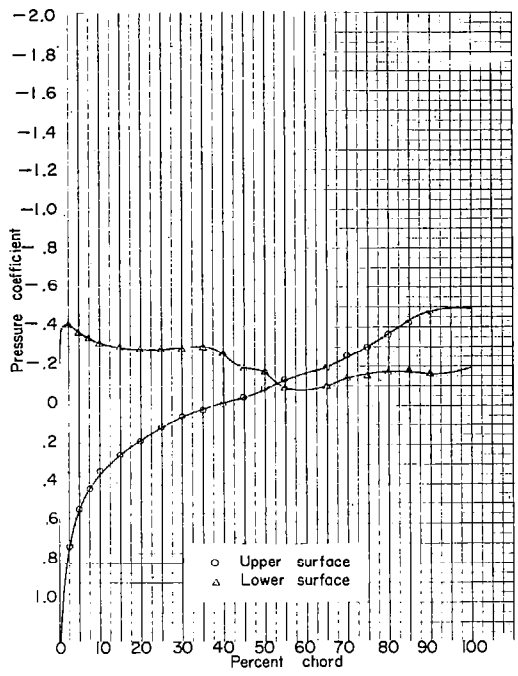
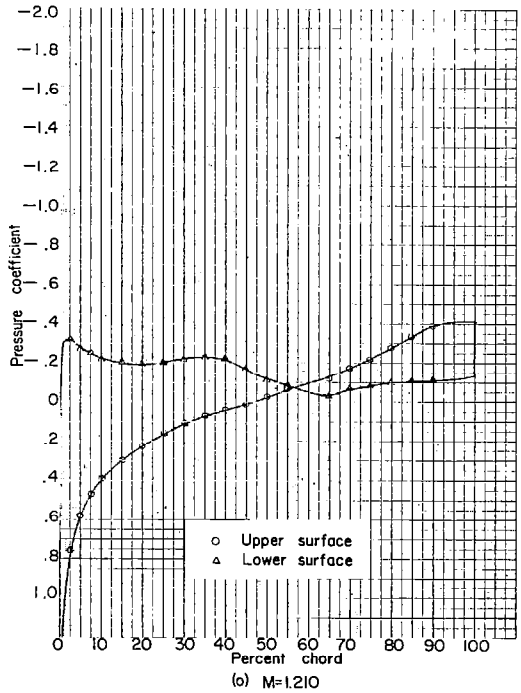


Figure 49.- Continued. NACA 16-506; $\alpha = -2^{\circ}$.







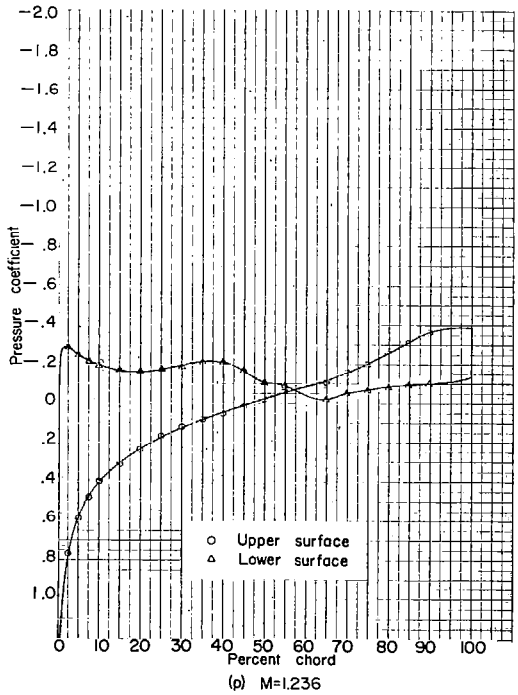


Figure 49.- Concluded. NACA 16-506; $\alpha = -2^{\circ}$.

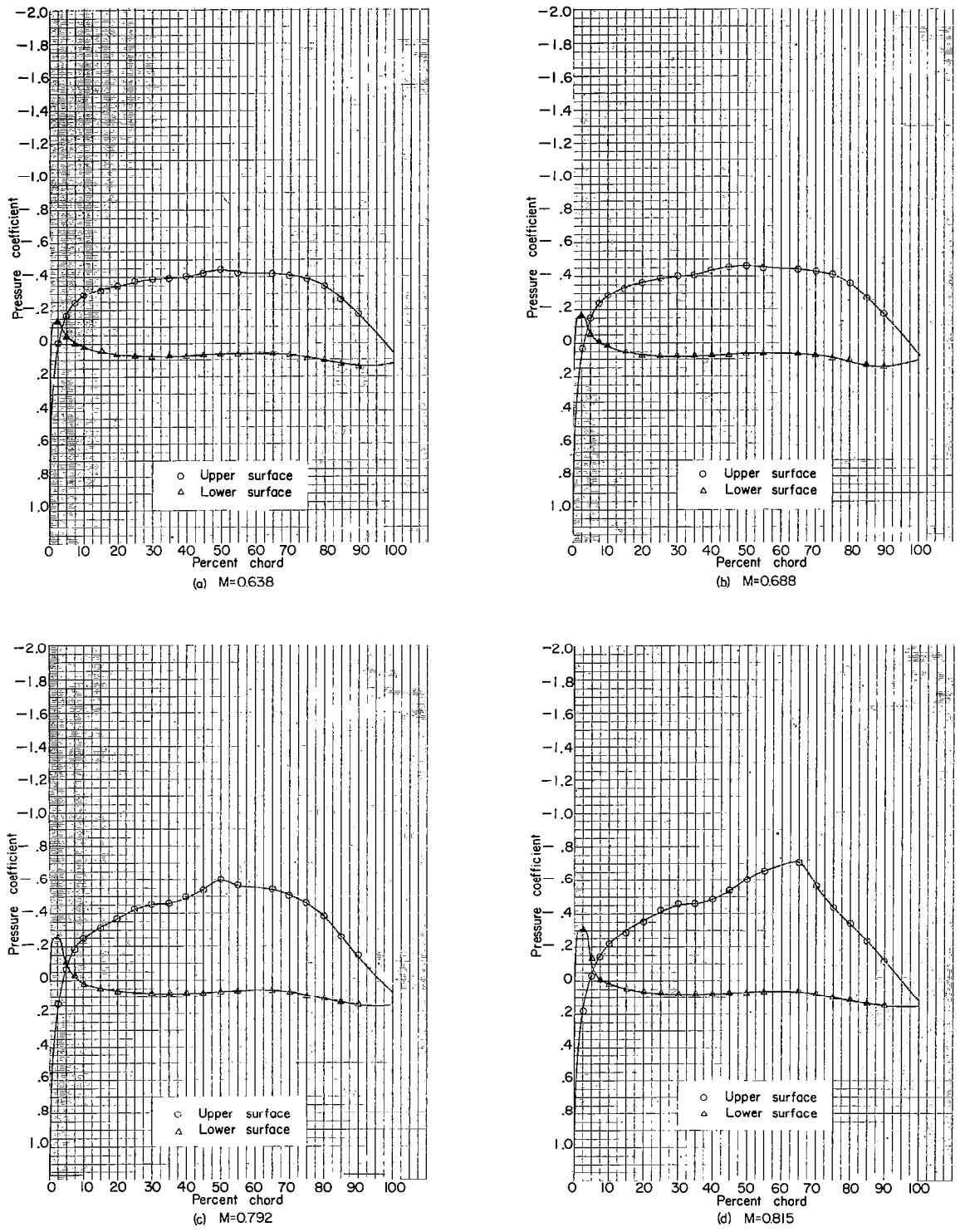
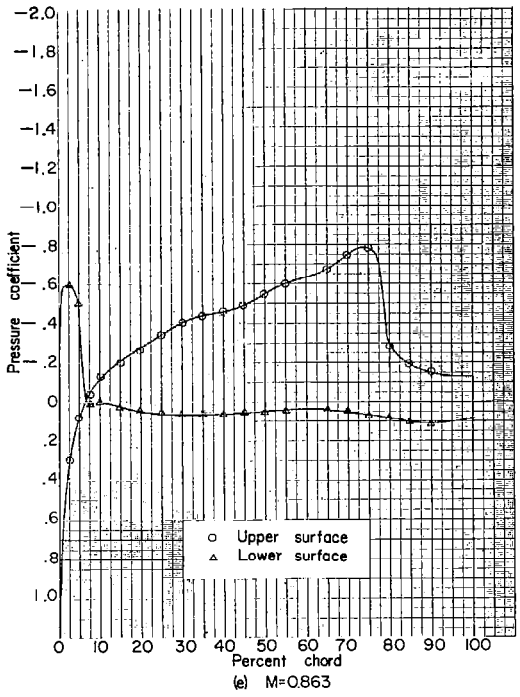
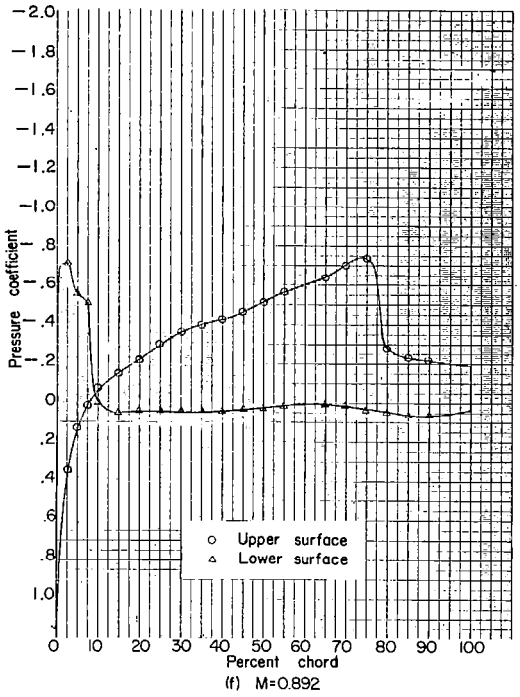
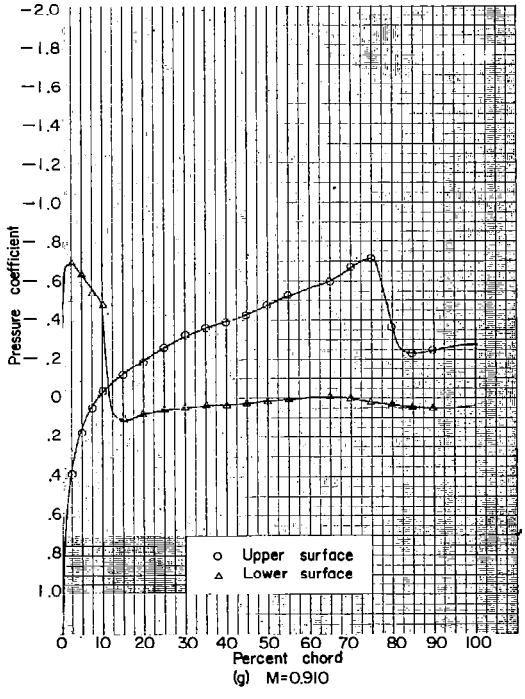


Figure 50.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 0^{\circ}$.







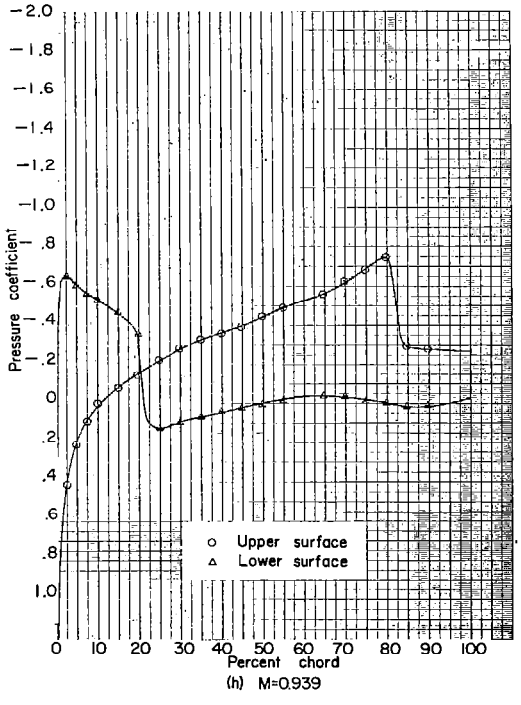
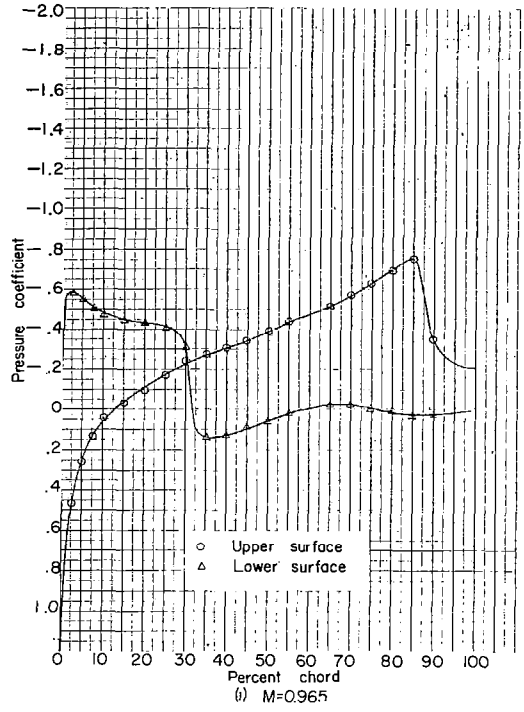
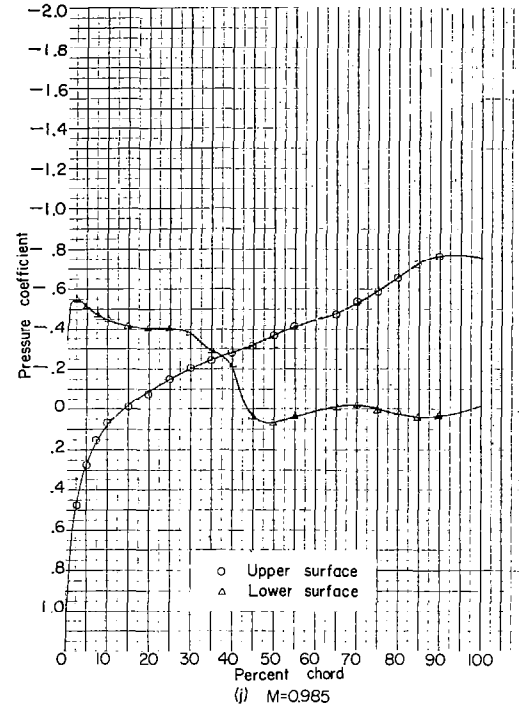
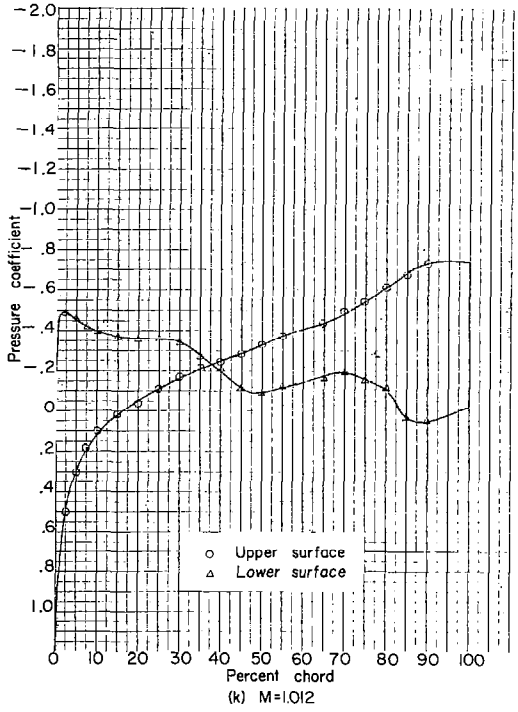


Figure 50.- Continued. NACA 16-506; $\alpha = 0^{\circ}$.







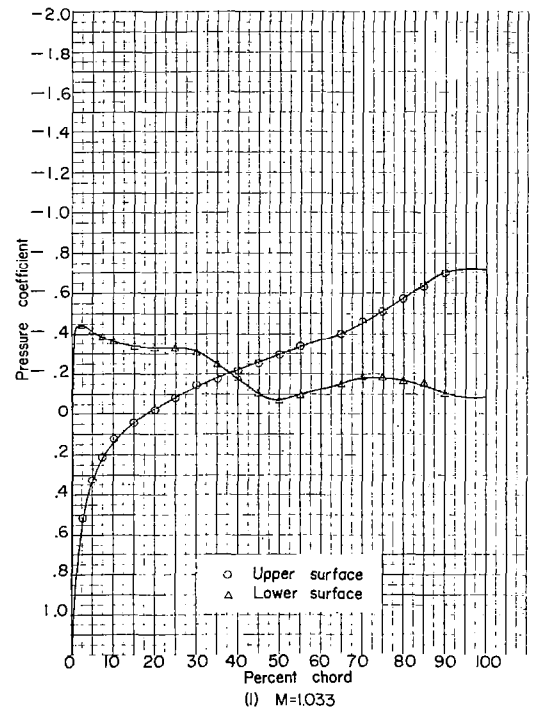


Figure 50.- Continued. NACA 16-506; $\alpha = 0^{\circ}$.

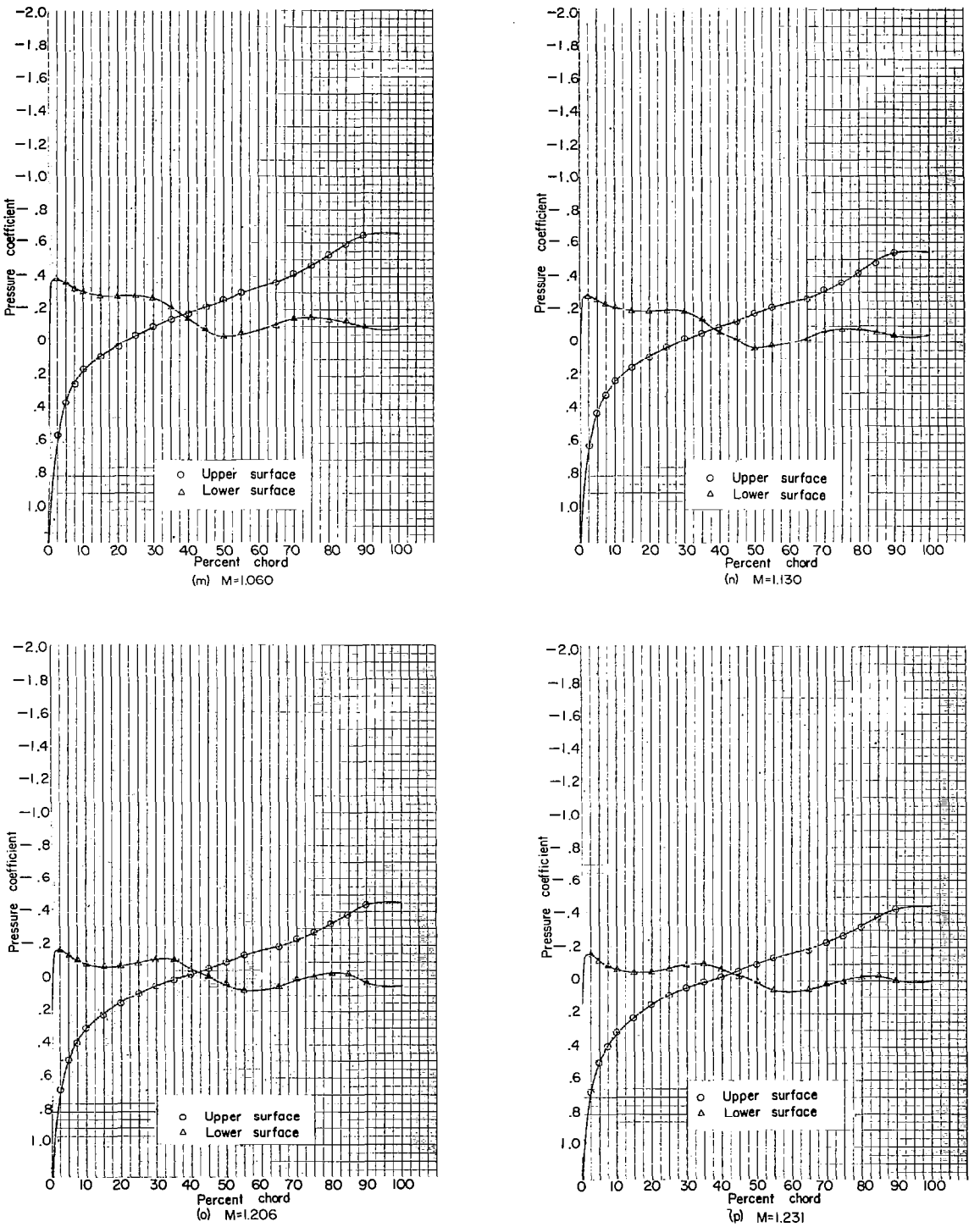


Figure 50.- Concluded. NACA 16-506; $\alpha = 0^{\circ}$.

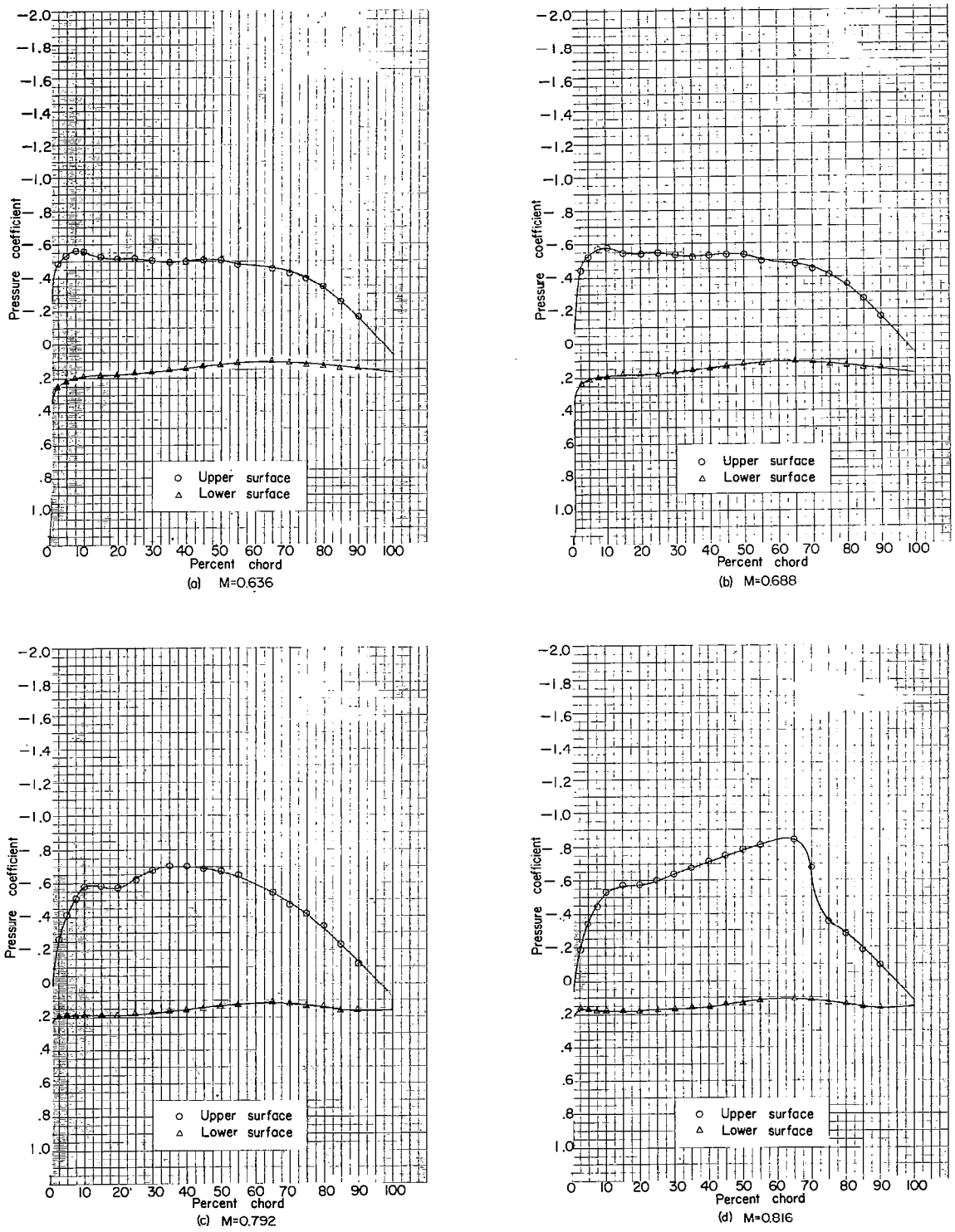
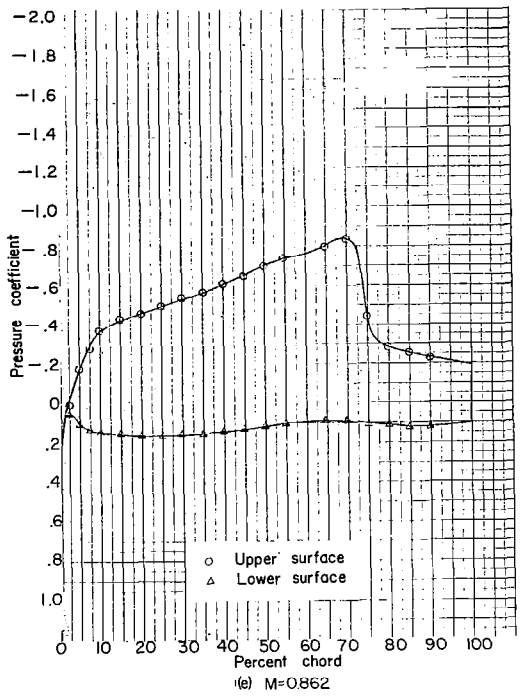
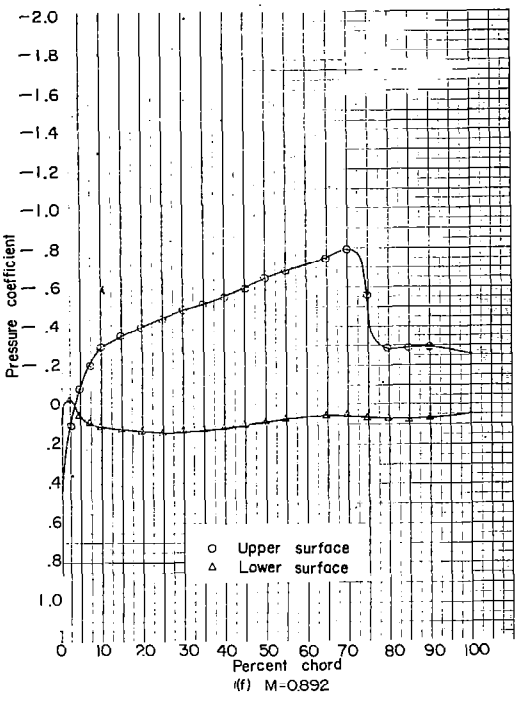


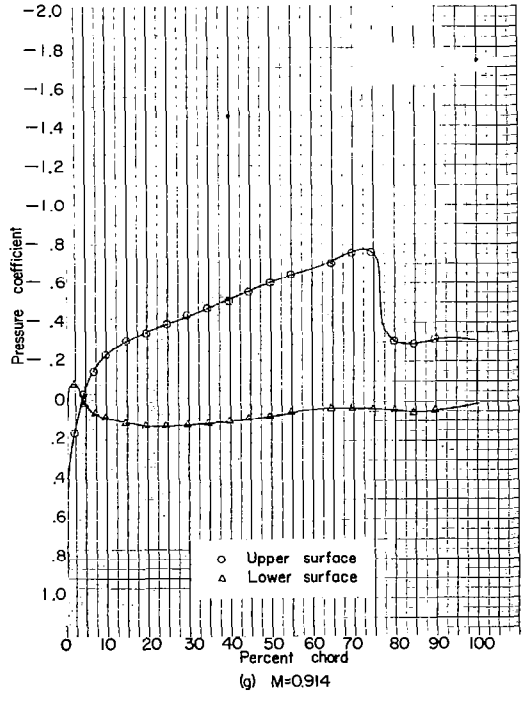
Figure 51.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 2^{\circ}$.



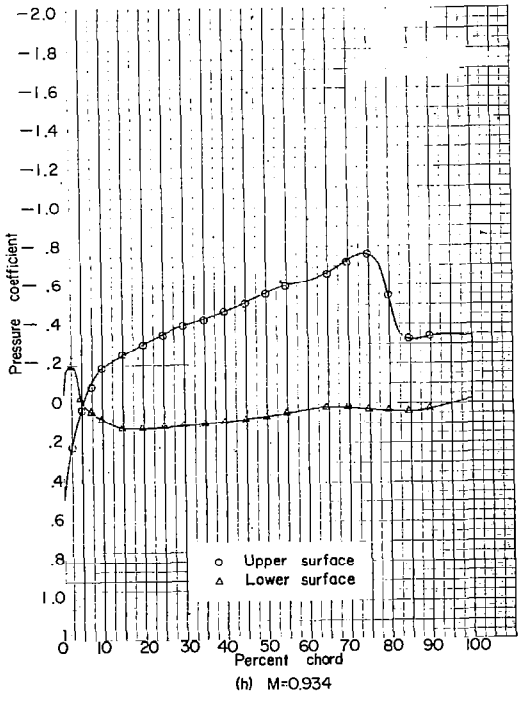
(e) M=0.862



(f) M=0.892

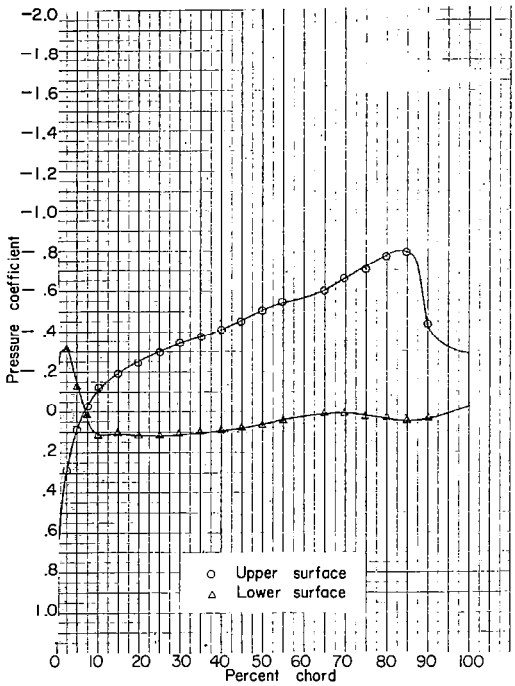


(g) M=0.914

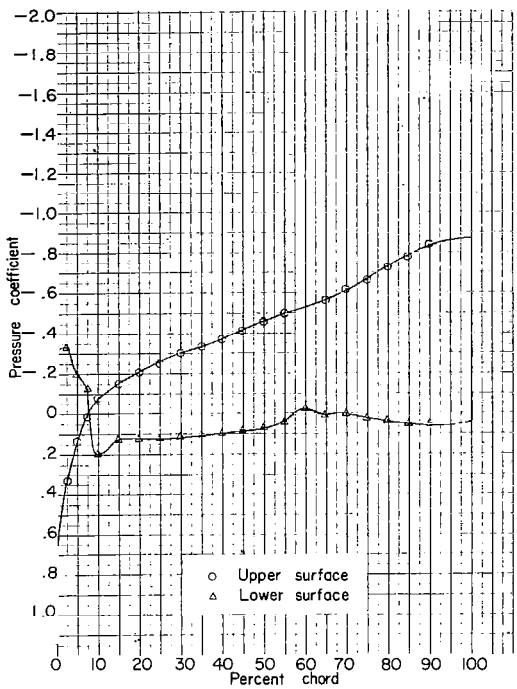


(h) M=0.934

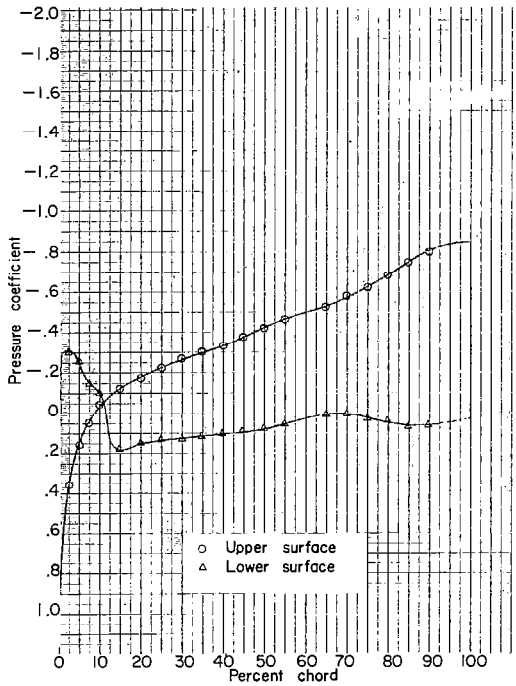
Figure 51.- Continued. NACA 16-506; $\alpha = 2^{\circ}$.



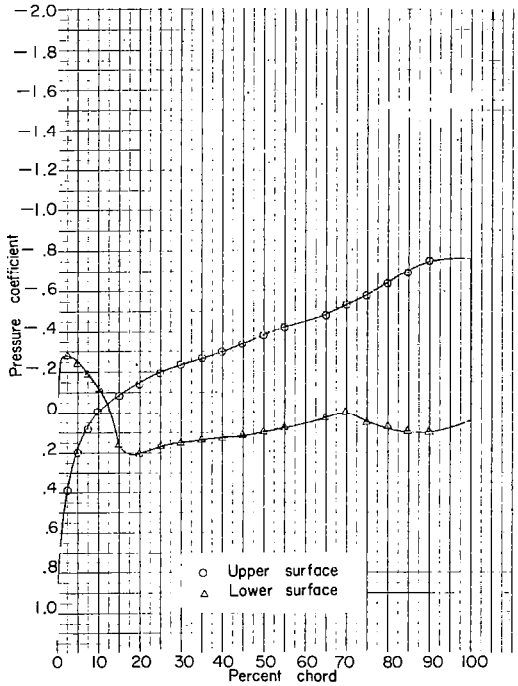
(i) M=0.962



(j) M=0.989

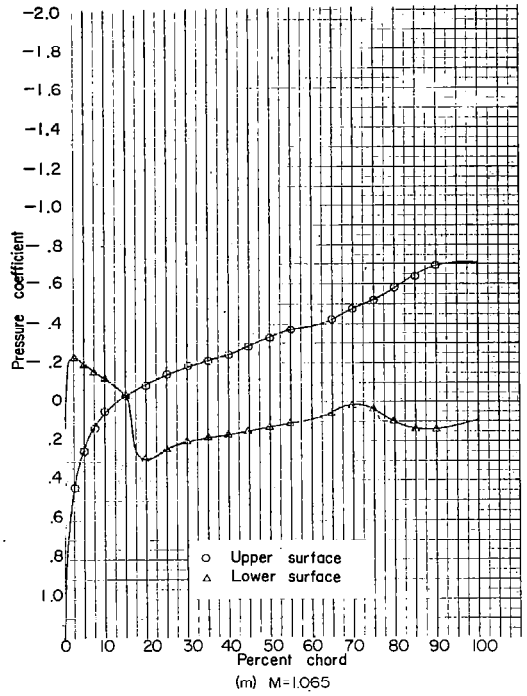


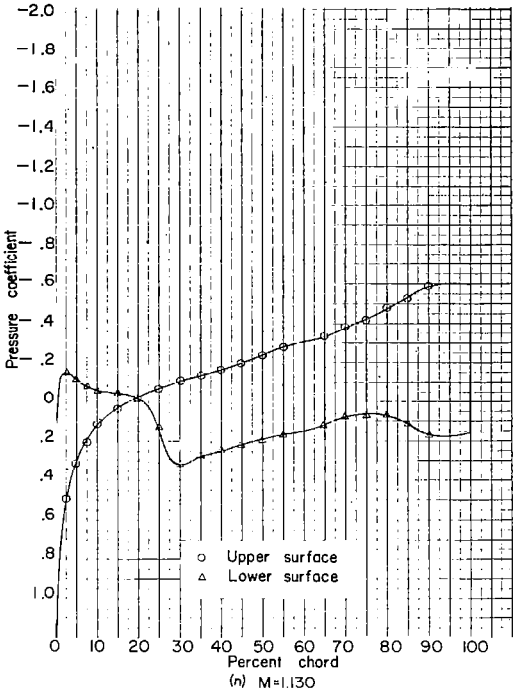
(k) M=1.009

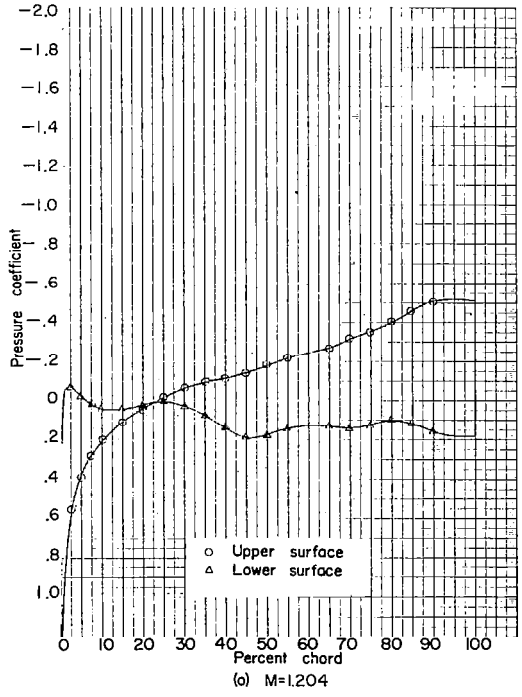


(l) M=1.035

Figure 51.- Continued. NACA 16-506; $\alpha = 2^{\circ}$.







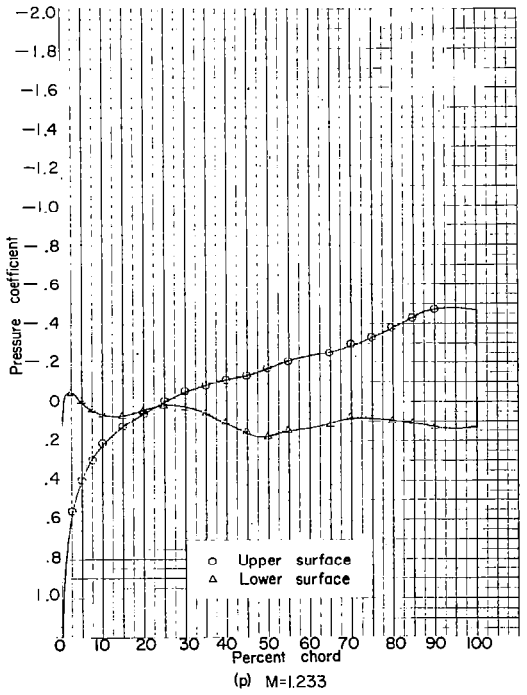


Figure 51.- Concluded. NACA 16-506; $\alpha = 2^{\circ}$.

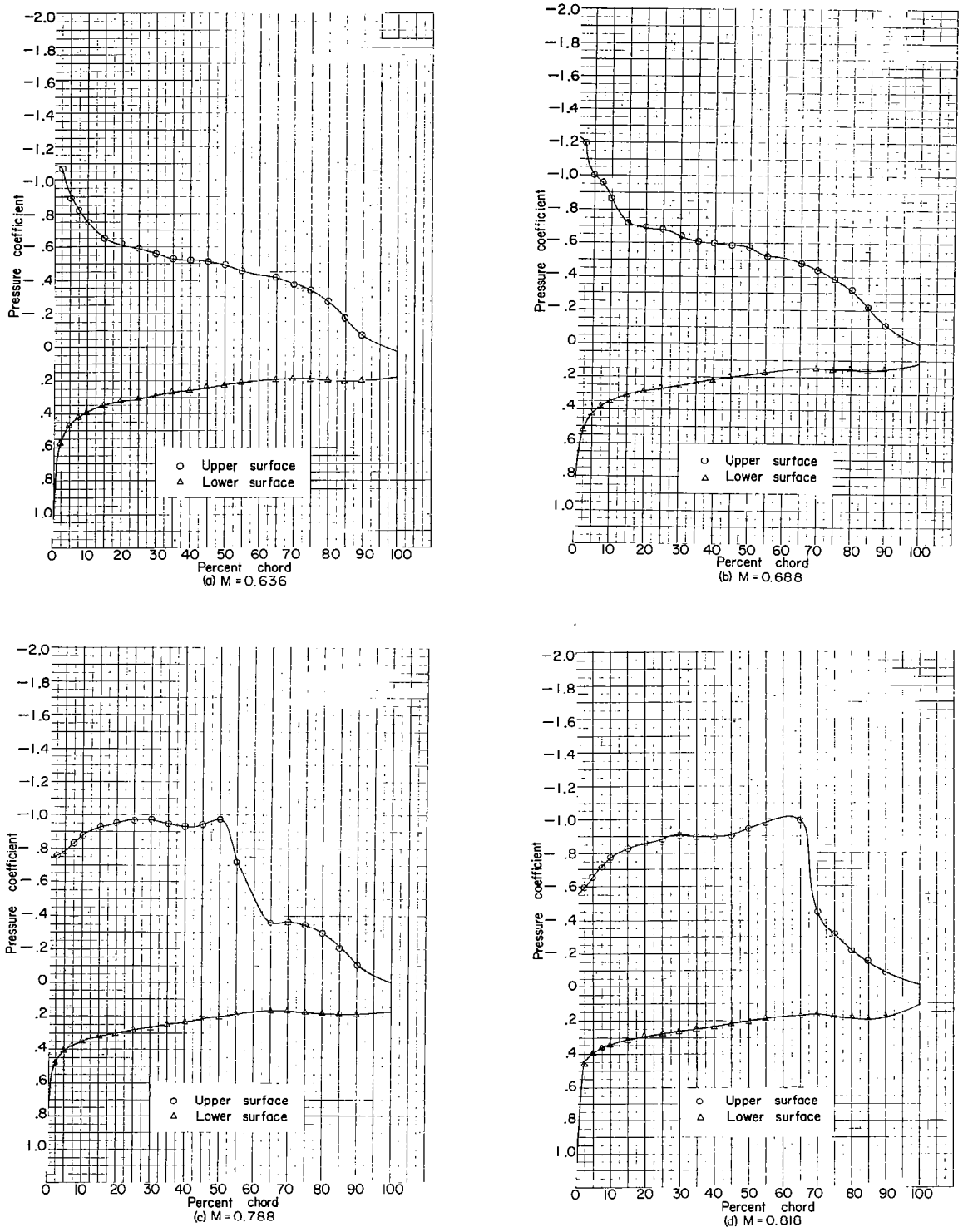
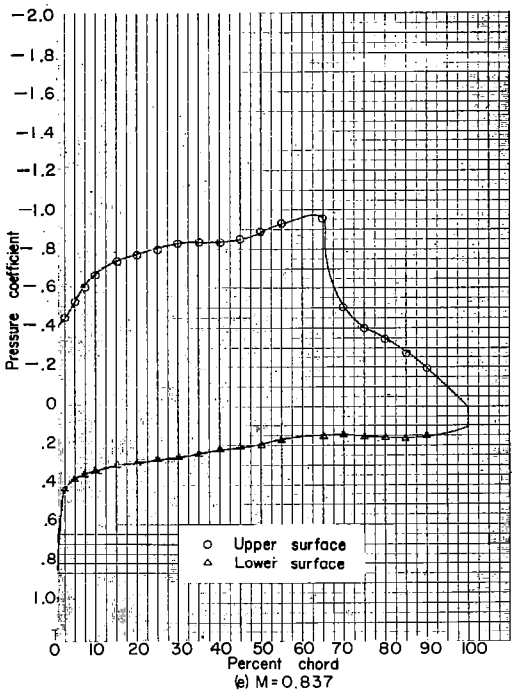
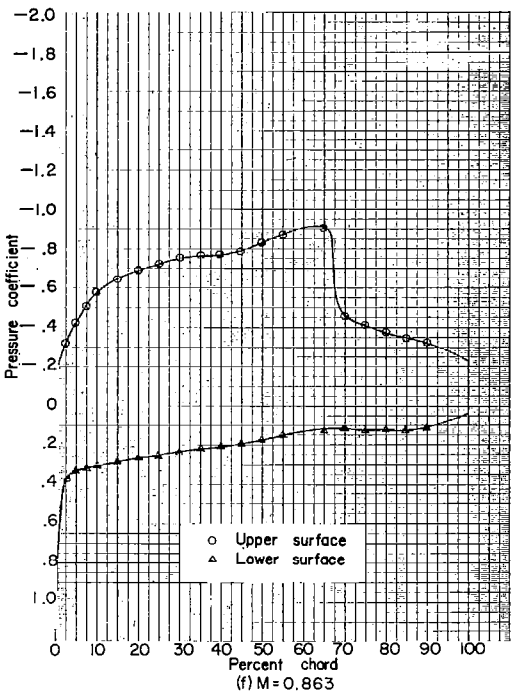
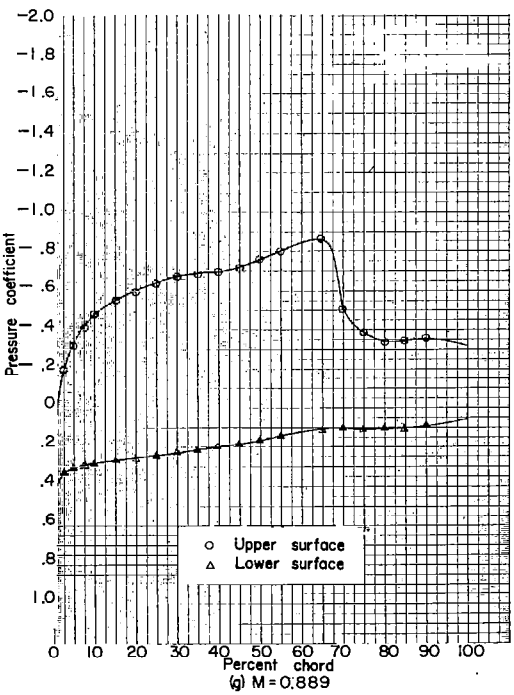


Figure 52.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 4^{\circ}$.







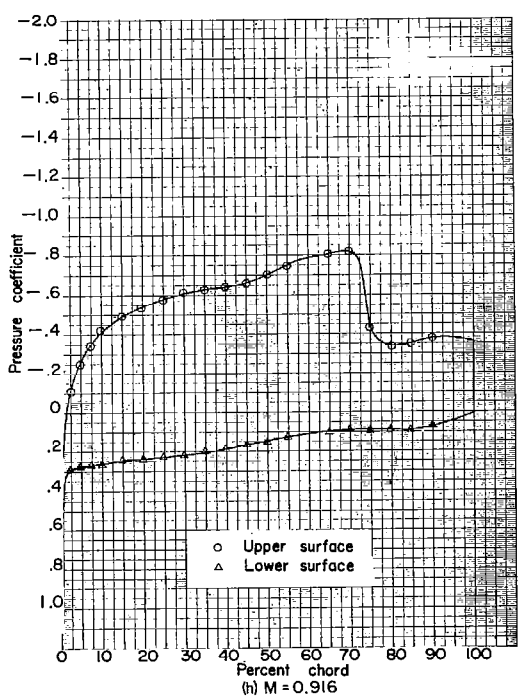
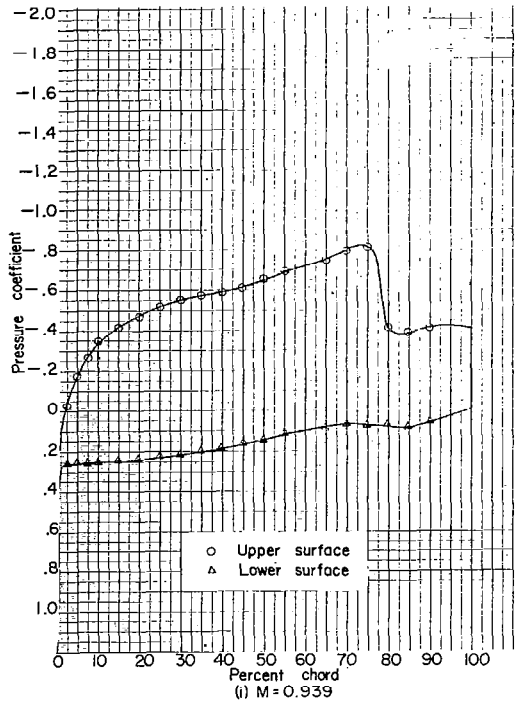
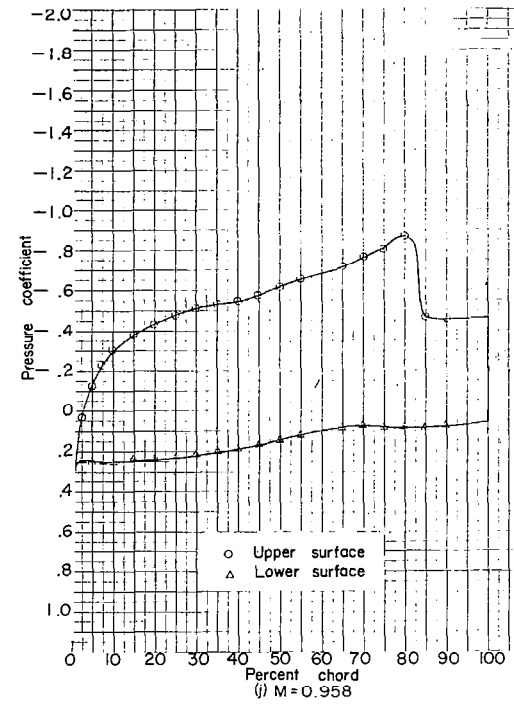
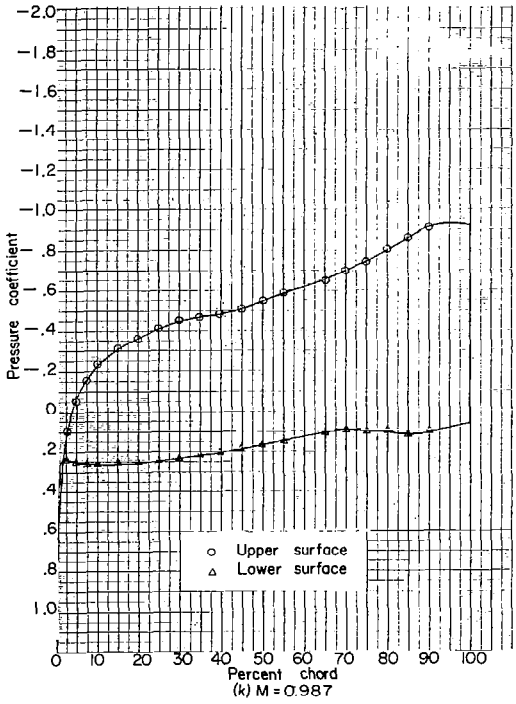


Figure 52.- Continued. NACA 16-506; $\alpha = 4^{\circ}$.







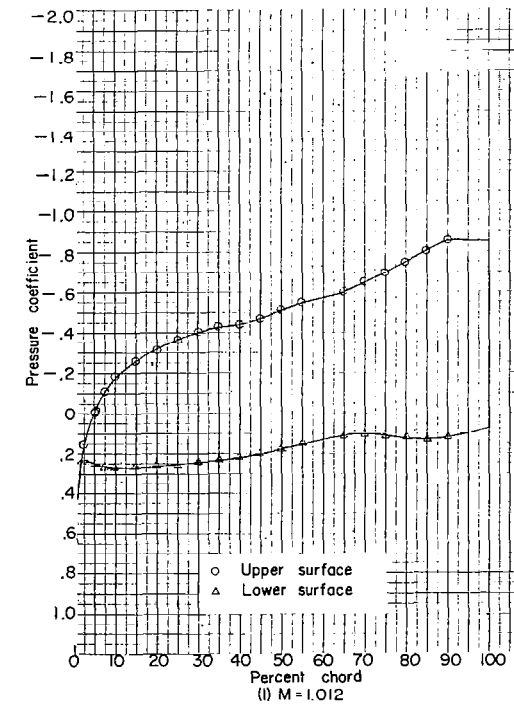
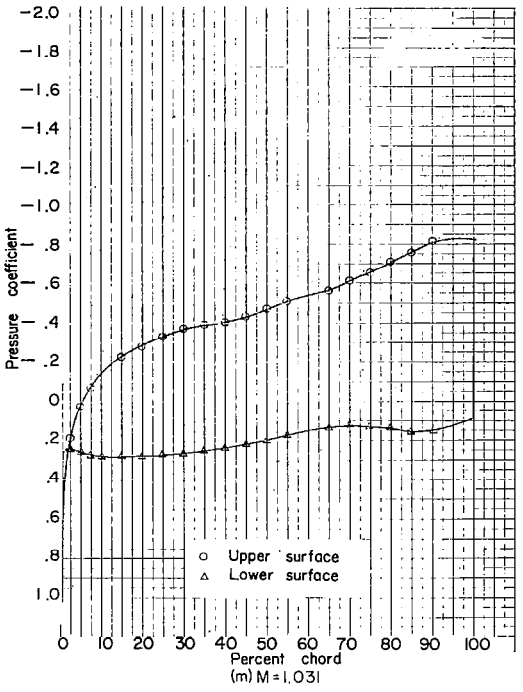
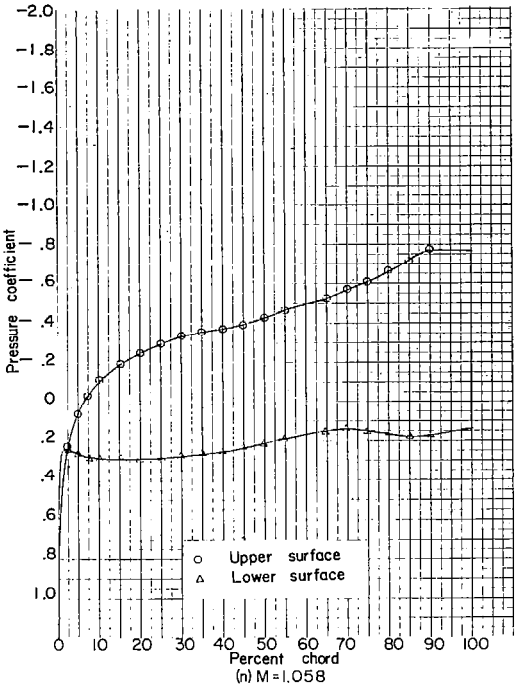
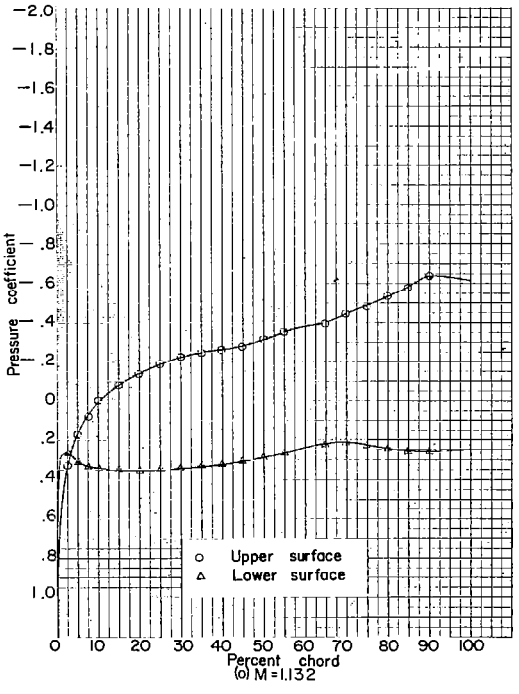


Figure 52.- Continued. NACA 16-506; $\alpha = 4^{\circ}$.







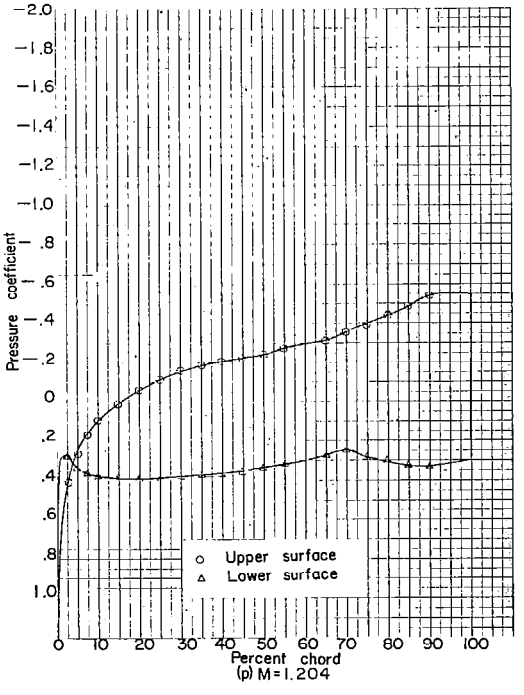


Figure 52.- Concluded. NACA 16-506; $\alpha = 4^{\circ}$.

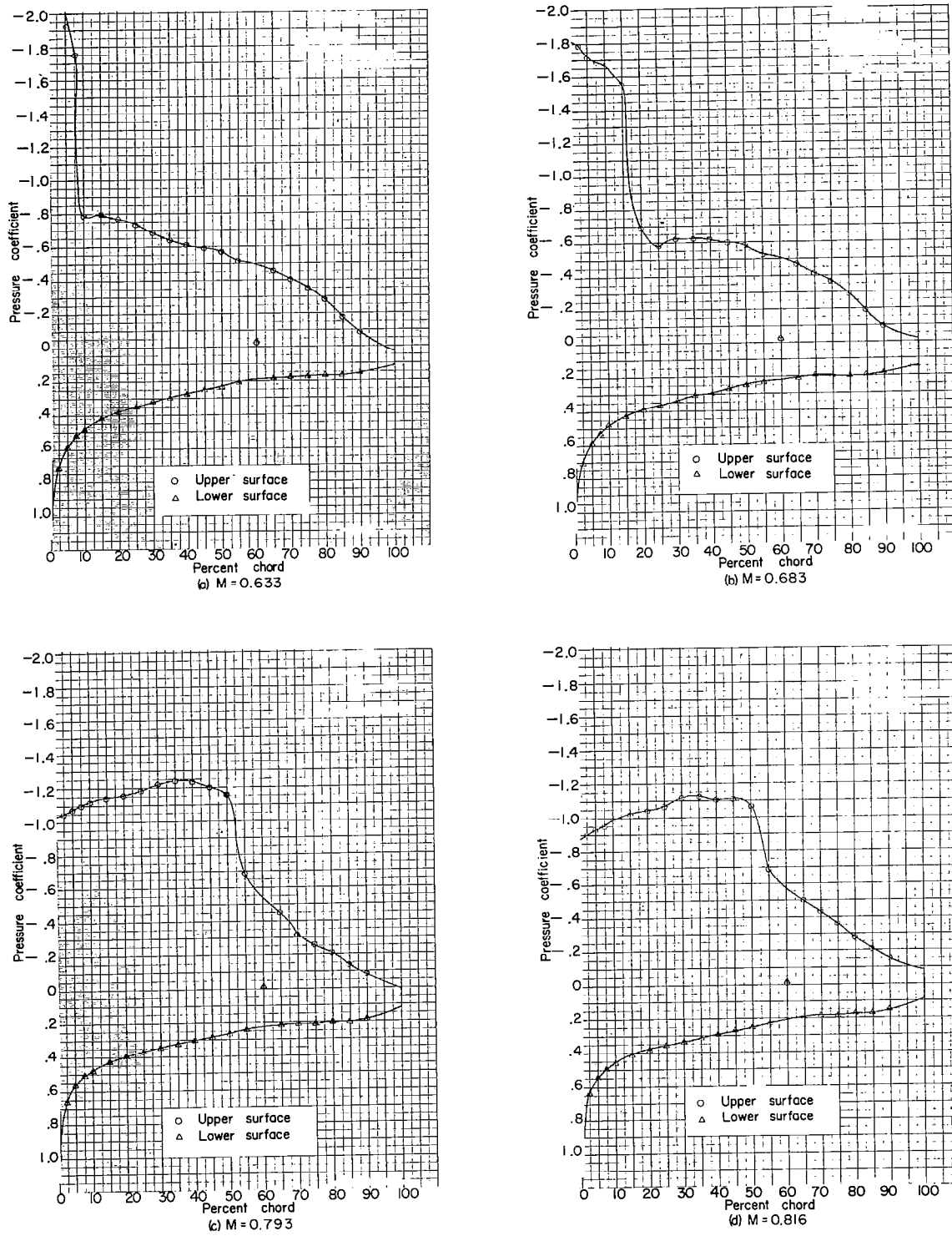


Figure 53.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 6^{\circ}$.

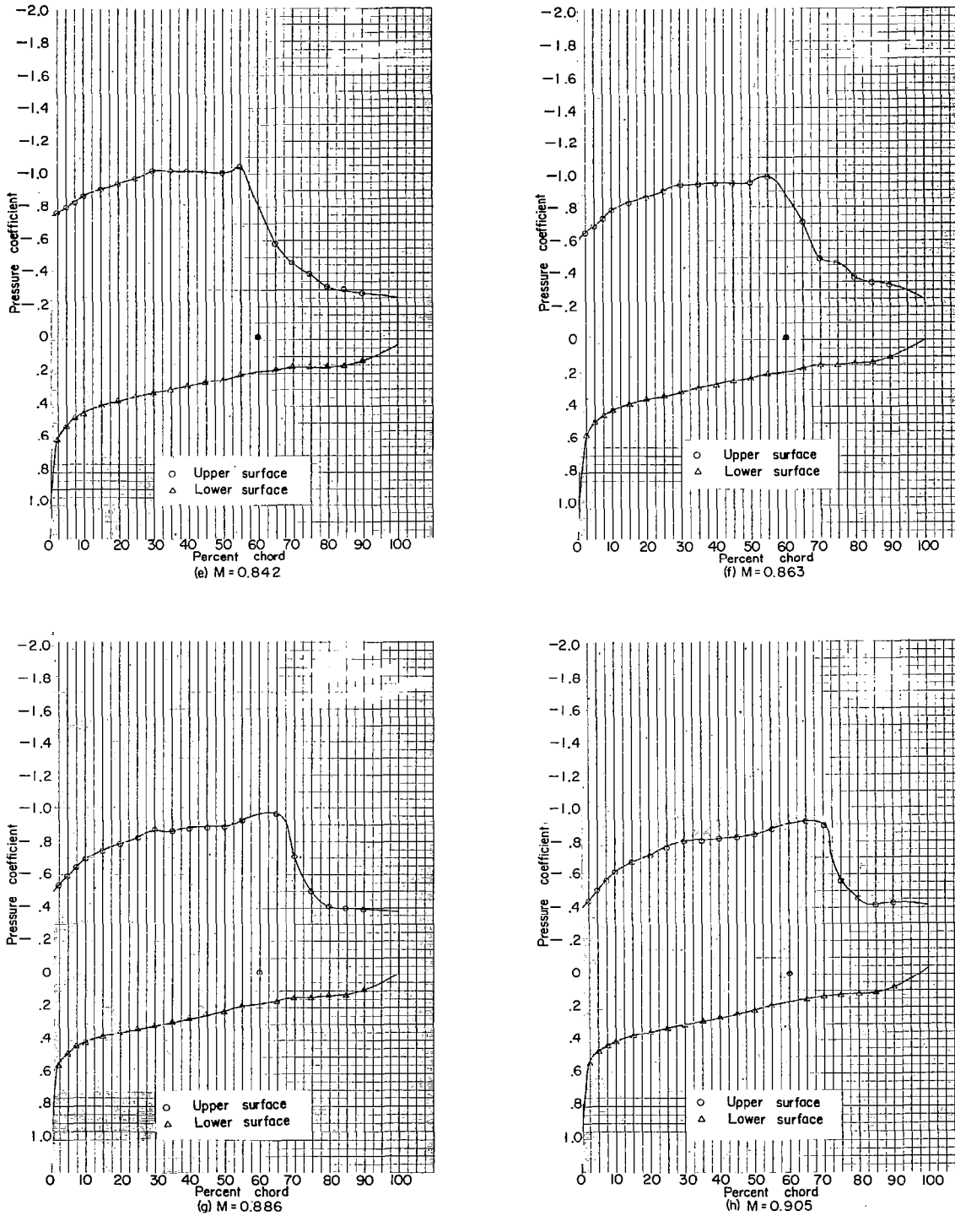
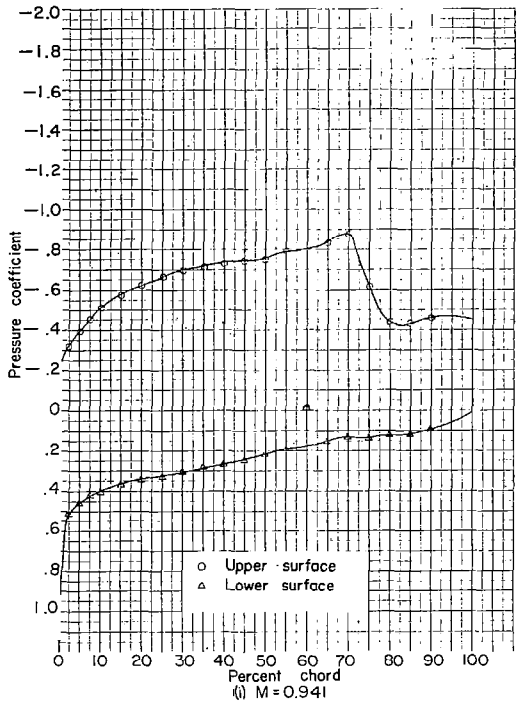
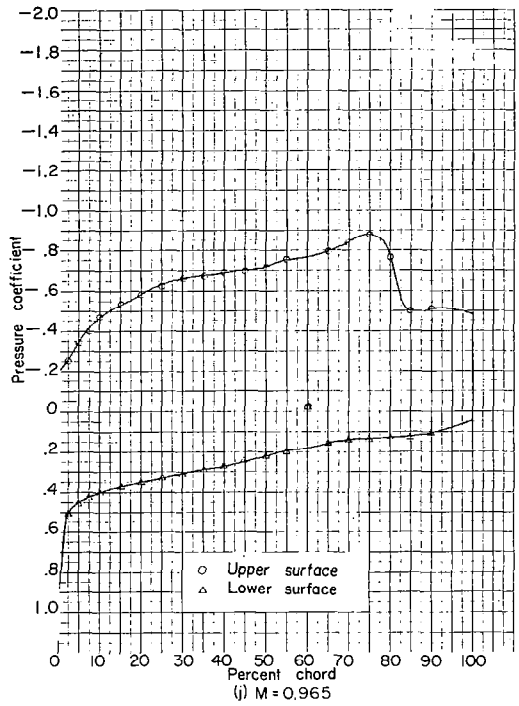
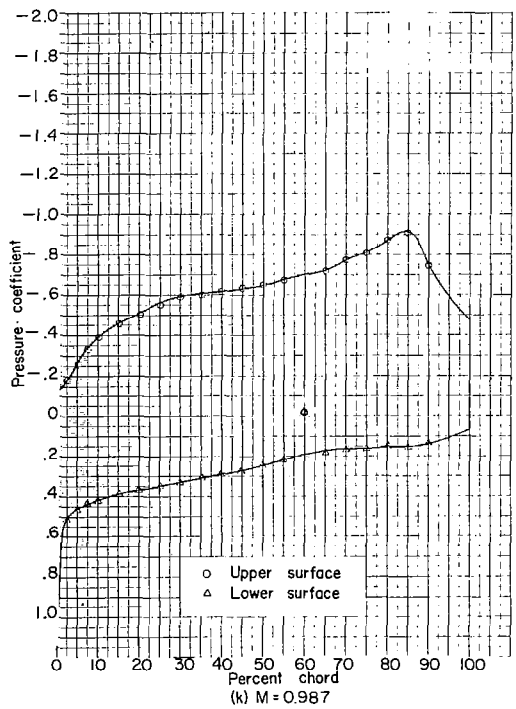


Figure 53.- Continued. NACA 16-506; $\alpha = 6^{\circ}$.







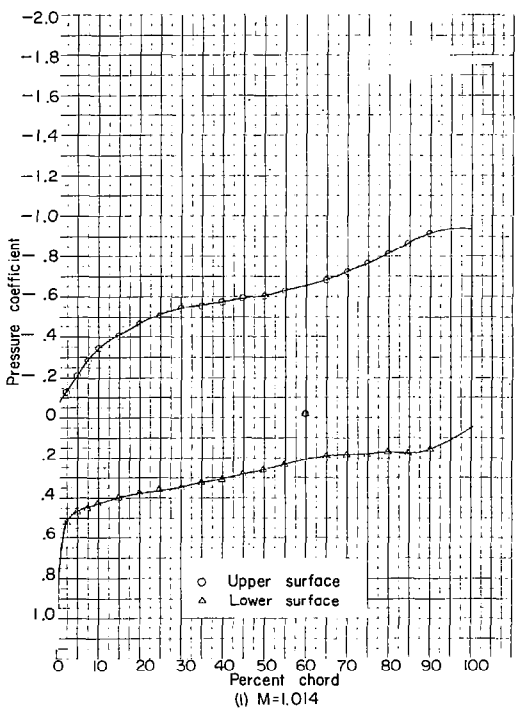
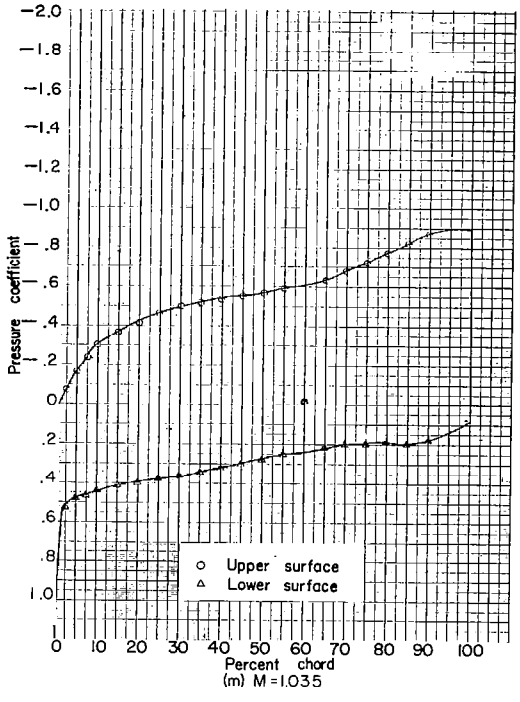
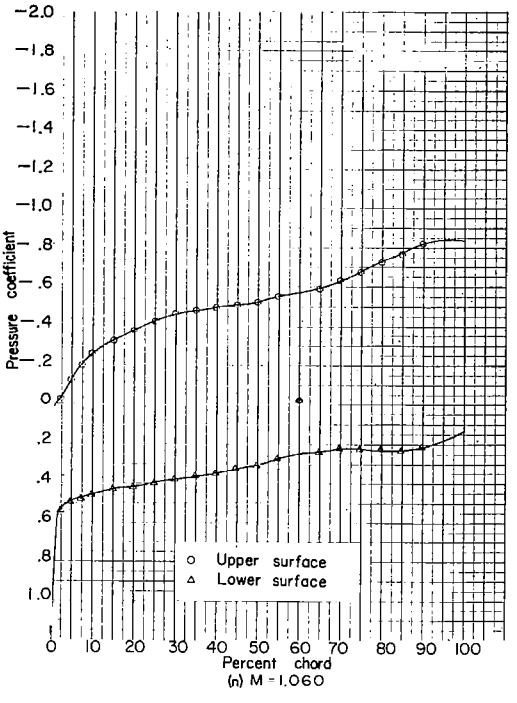
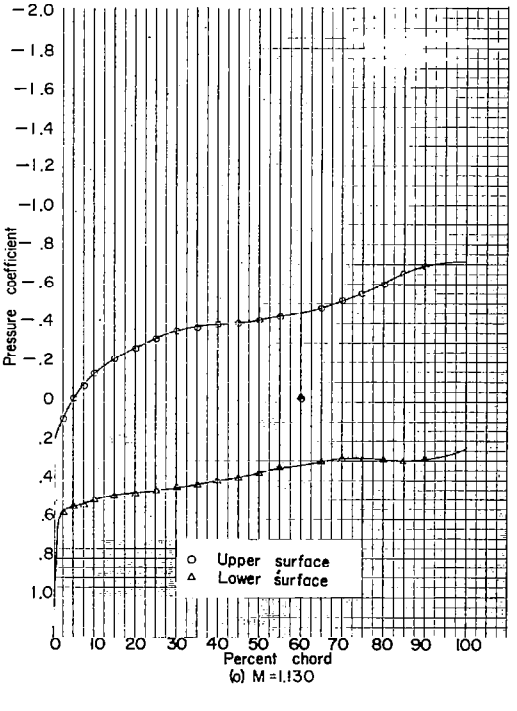


Figure 53.- Continued. NACA 16-506; $\alpha = 6^{\circ}$.







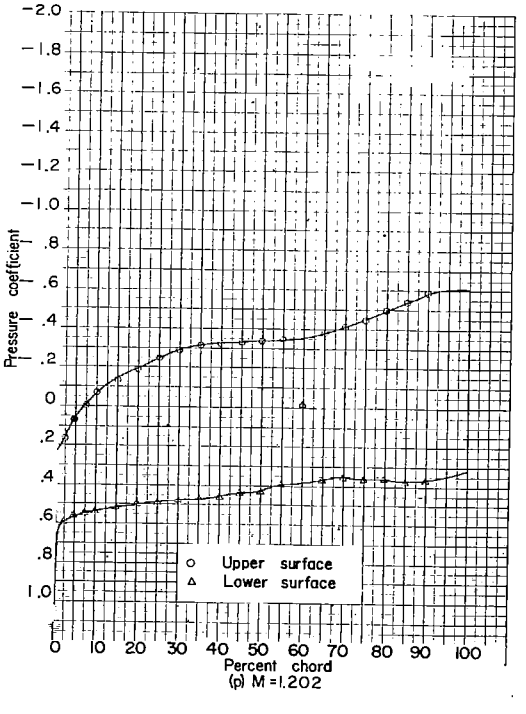


Figure 53.- Concluded. NACA 16-506; $\alpha = 6^{\circ}$.

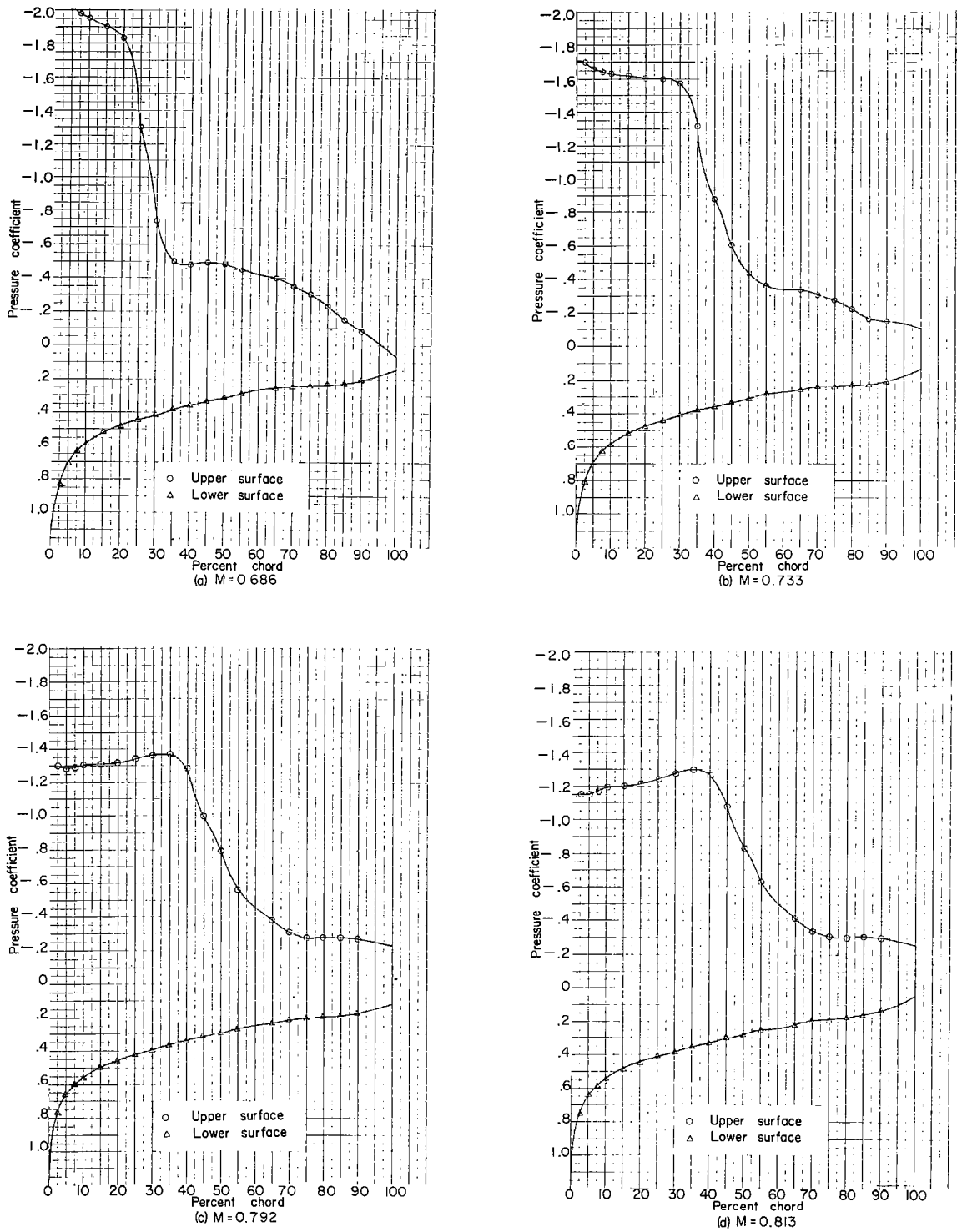
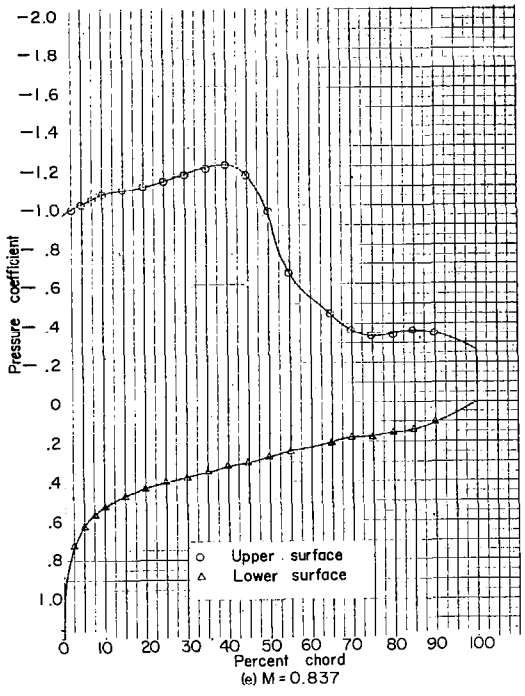
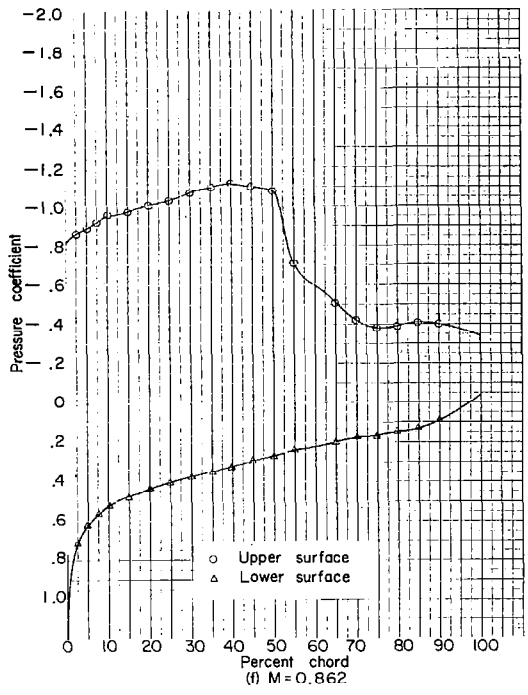
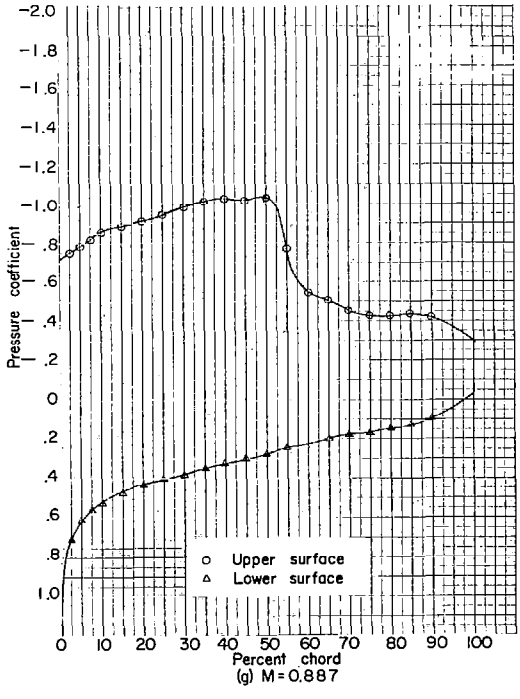


Figure 54.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 8^{\circ}$.







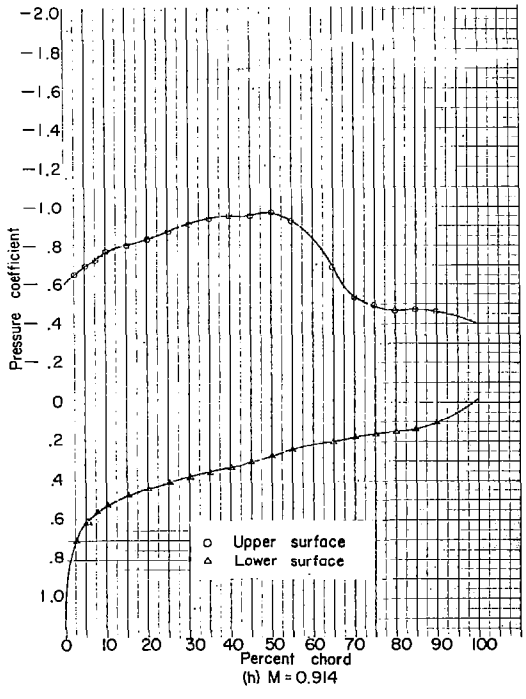
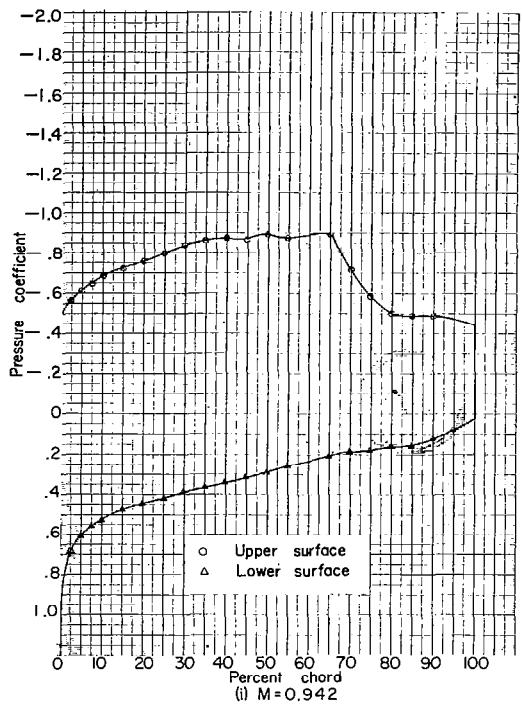
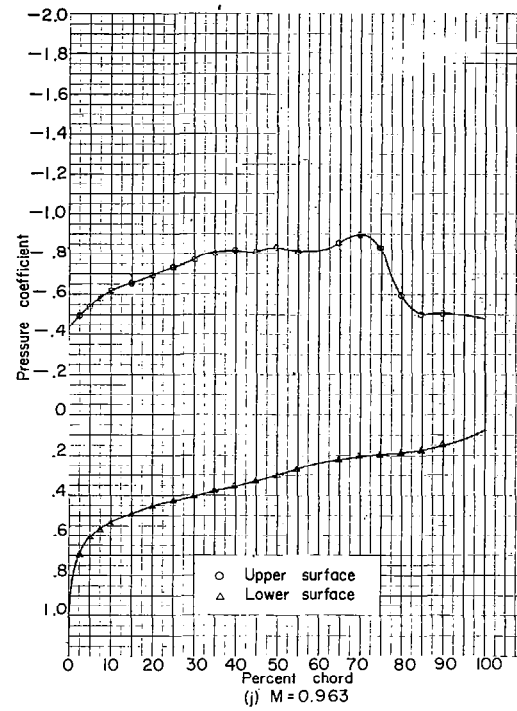
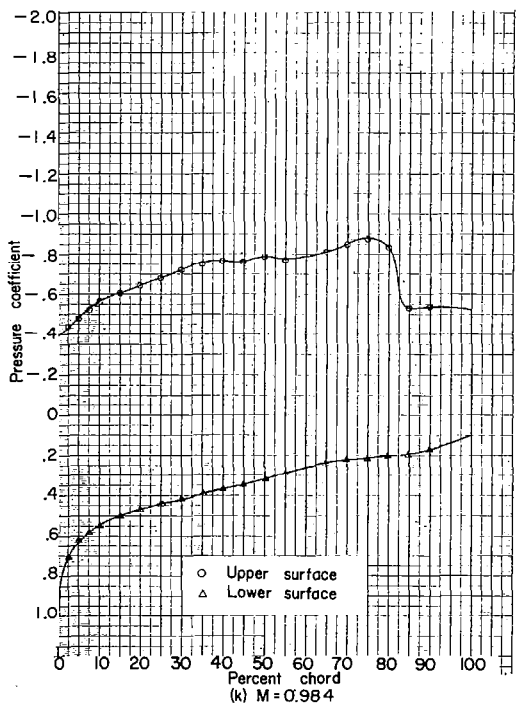


Figure 54.- Continued. NACA 16-506; $\alpha = 8^{\circ}$.







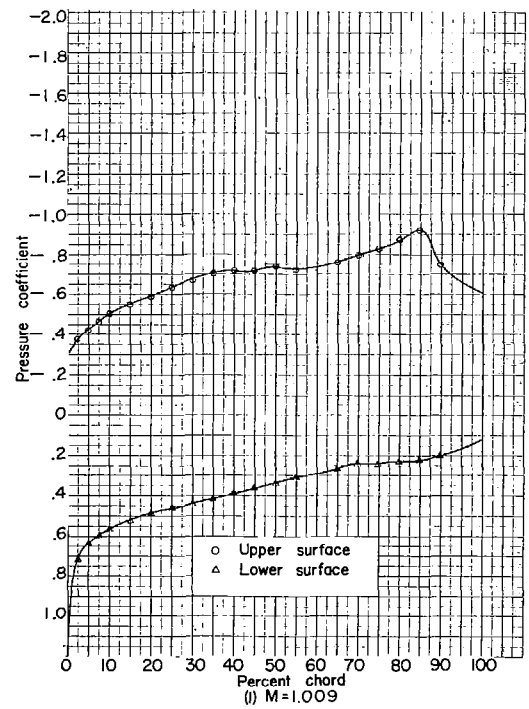
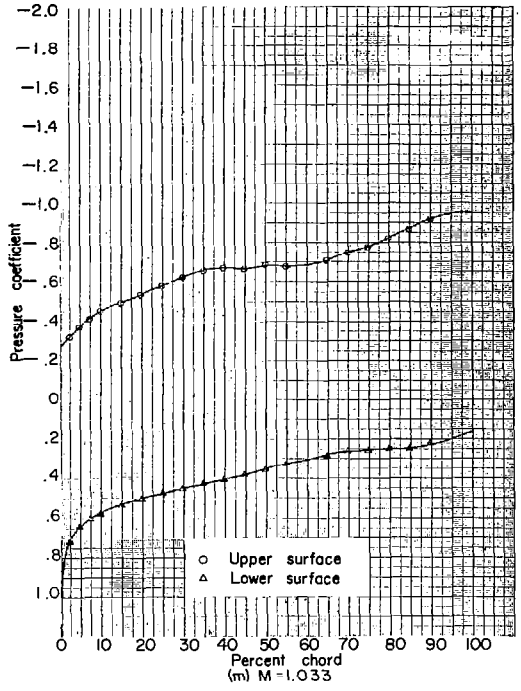
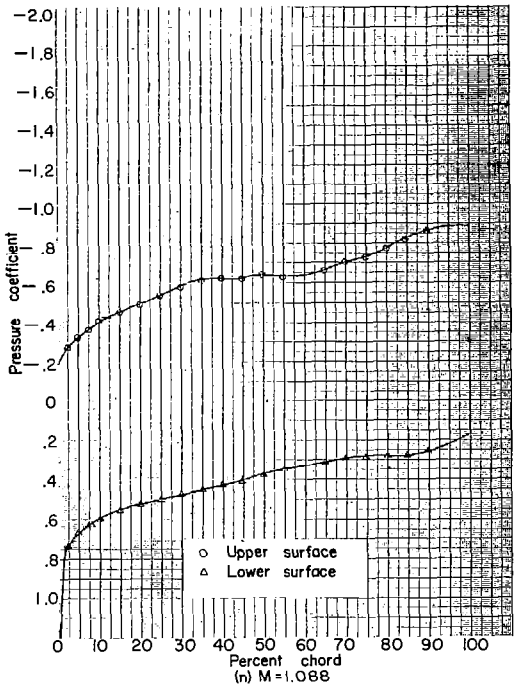
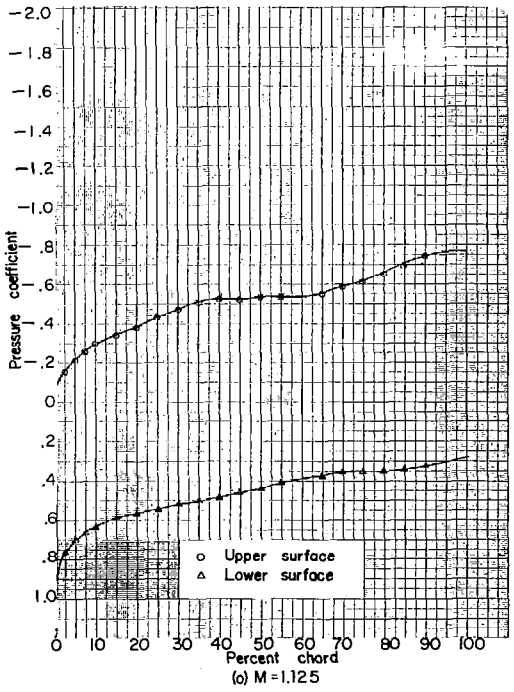


Figure 54.- Continued. NACA 16-506; $\alpha = 8^{\circ}$.







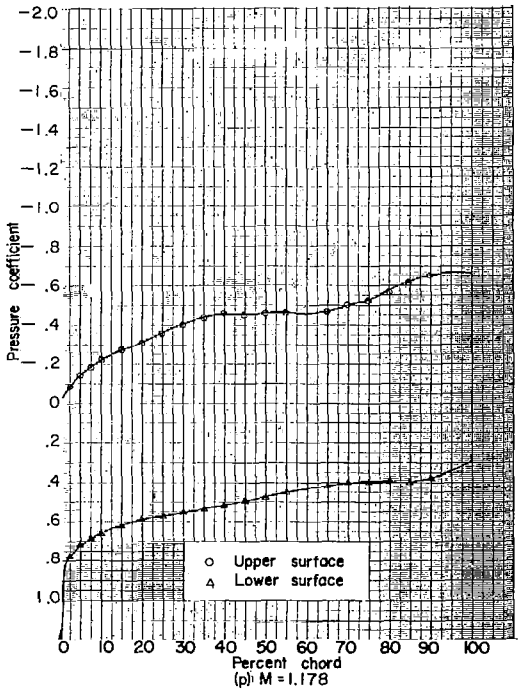


Figure 54.- Concluded. NACA 16-506; $\alpha = 8^{\circ}$.

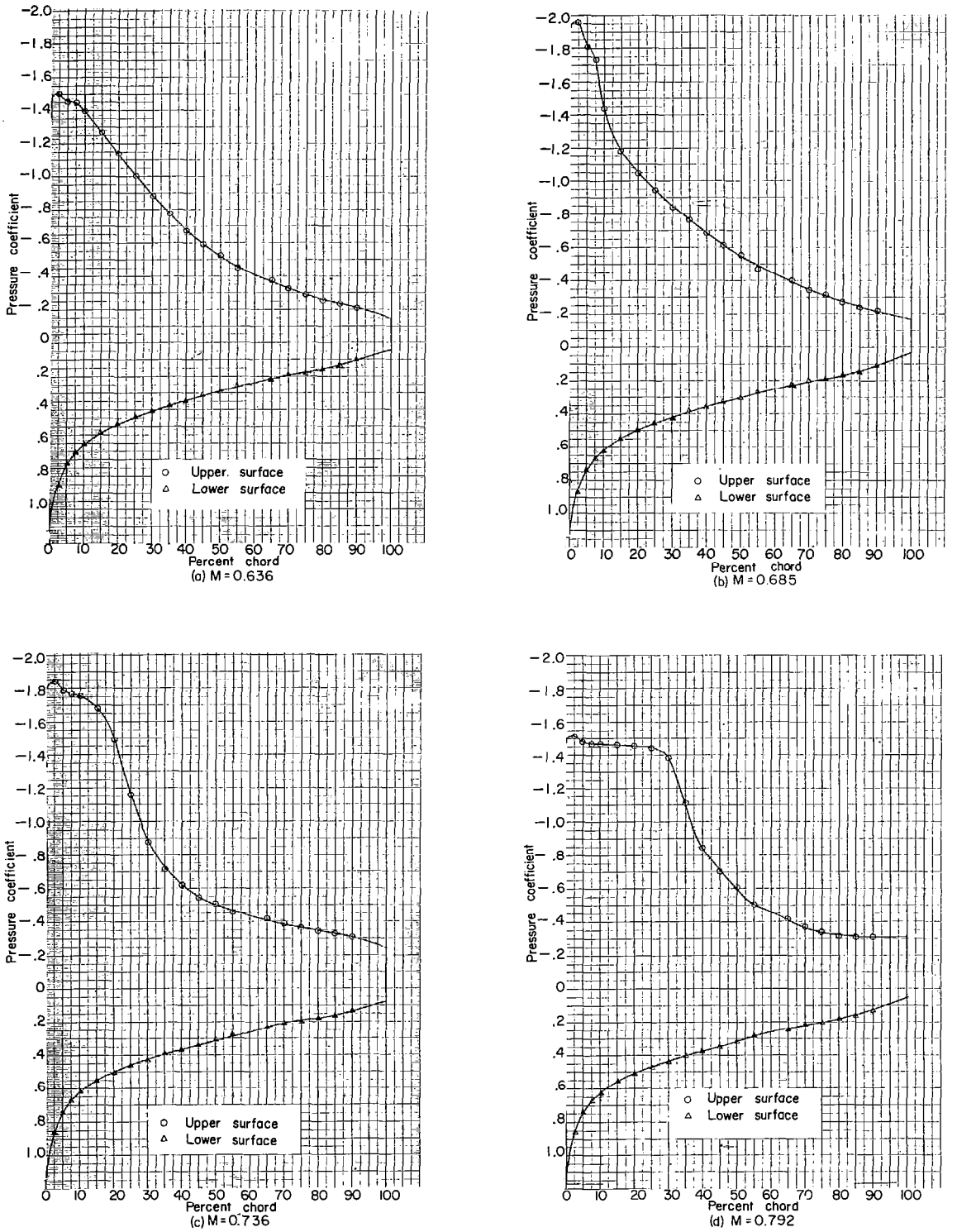
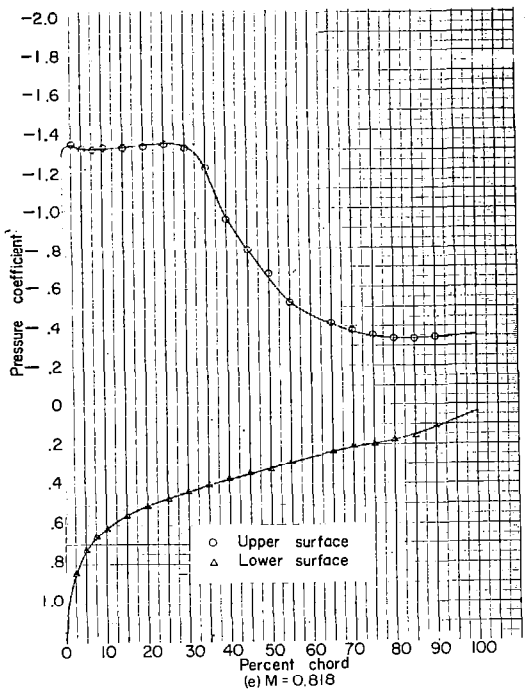
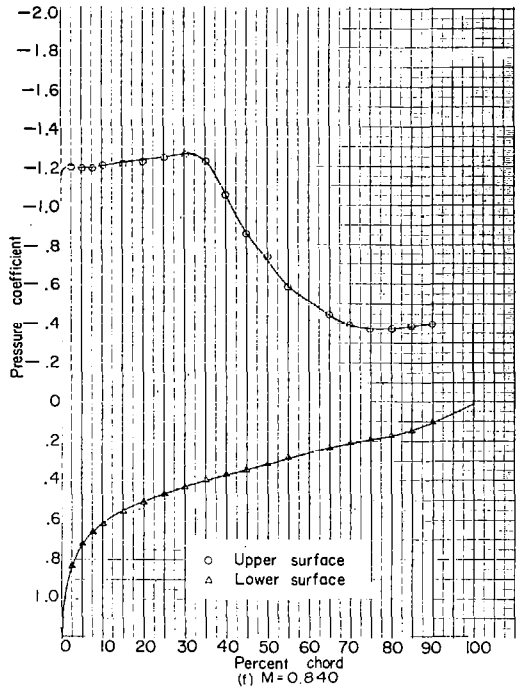
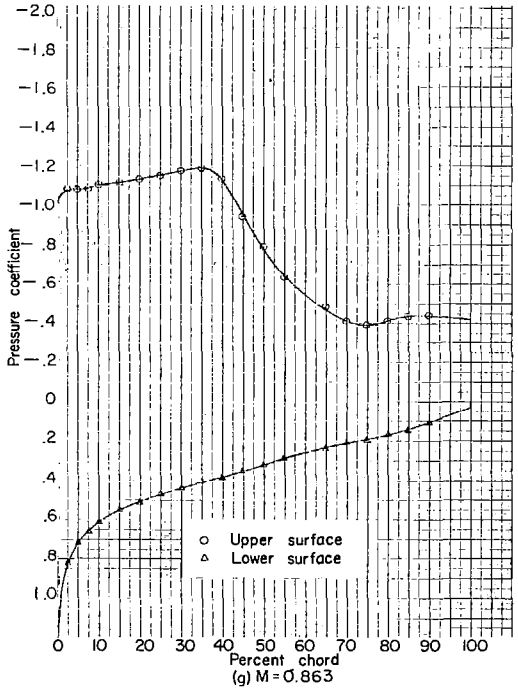


Figure 55.- Pressure distributions over NACA 16-506 airfoil section. $\alpha = 10^{\circ}$.







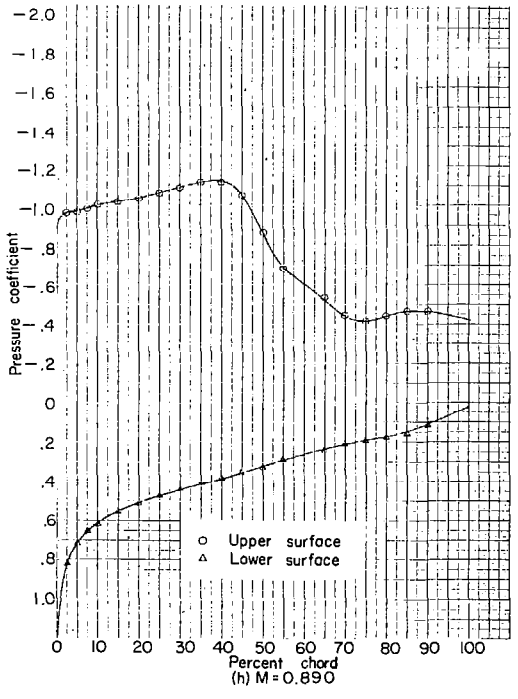
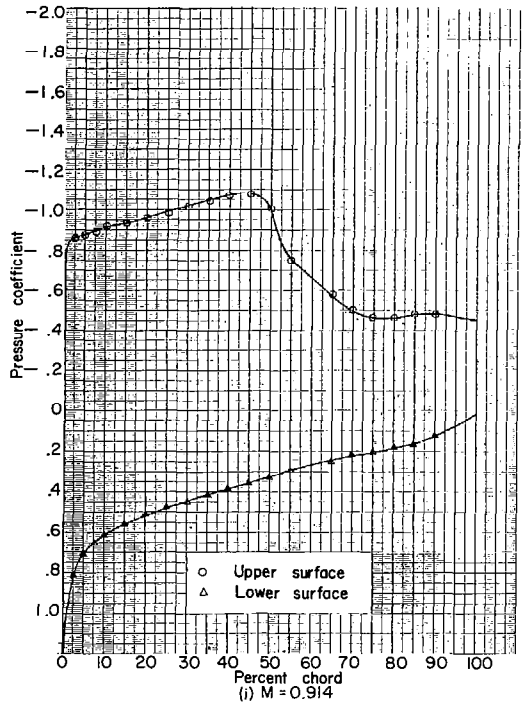
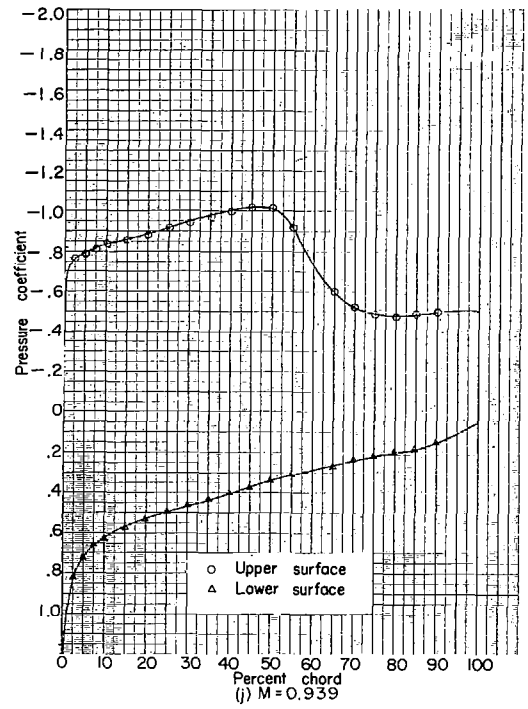
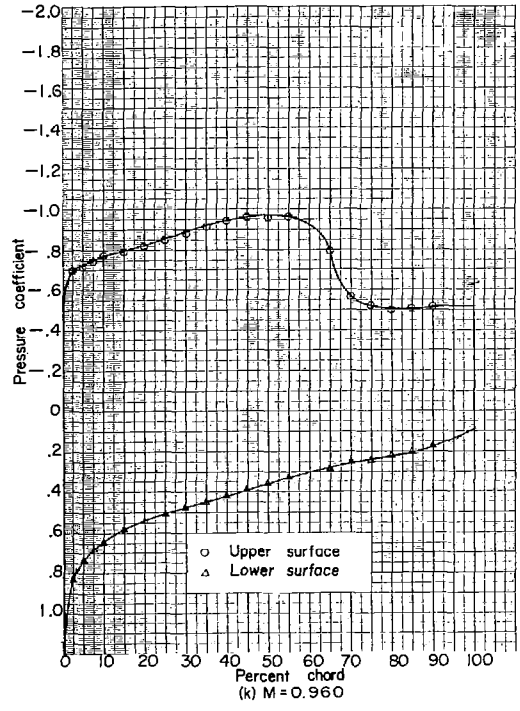


Figure 55.- Continued. NACA 16-506; $\alpha = 10^{\circ}$.







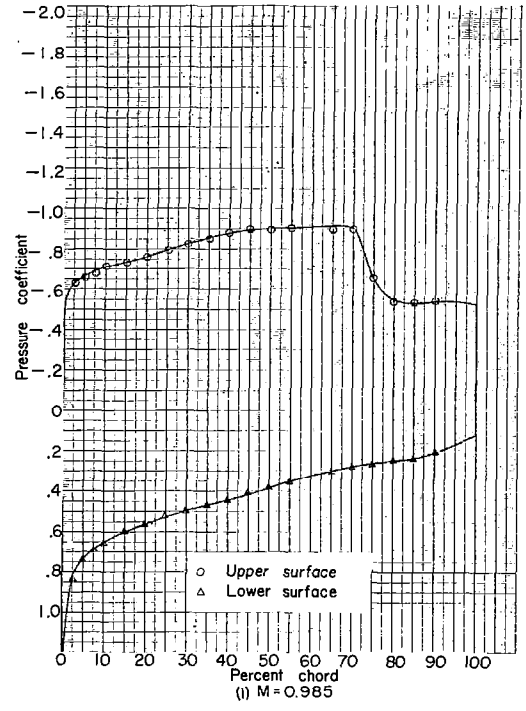
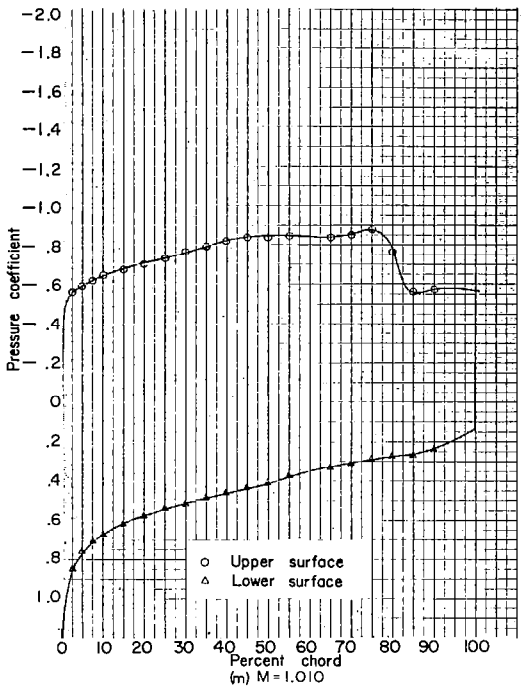
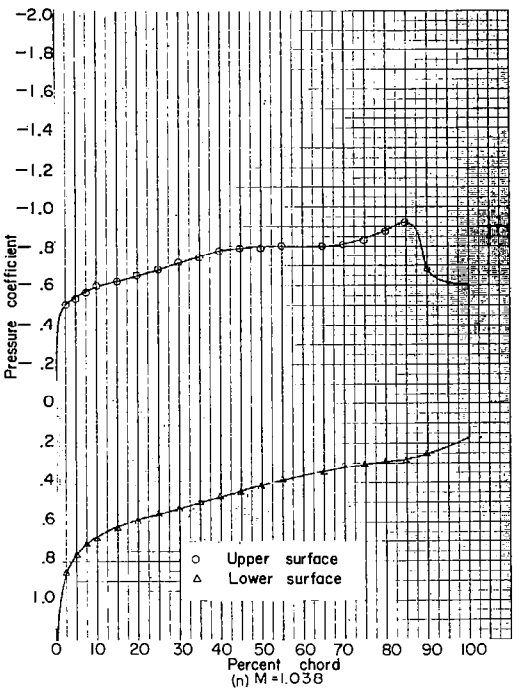
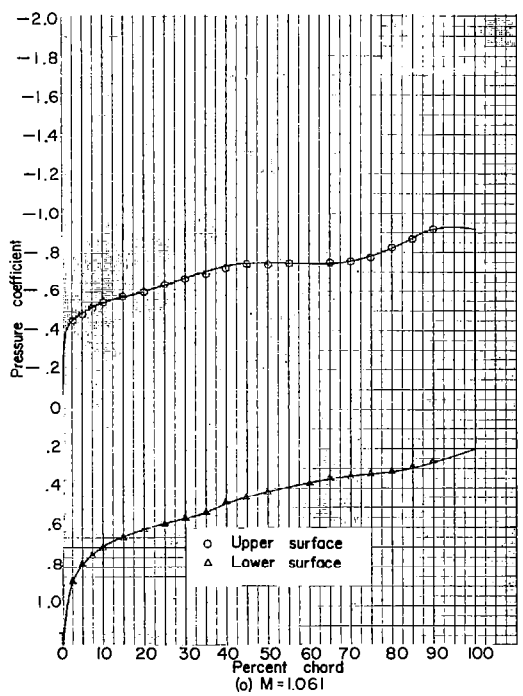


Figure 55.- Continued. NACA 16-506; $\alpha = 10^{\circ}$.







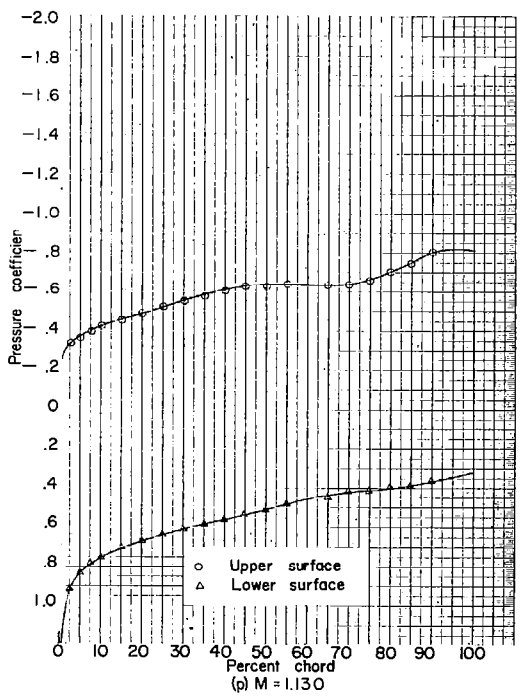


Figure 55.- Concluded. NACA 16-506; $\alpha = 10^{\circ}$.

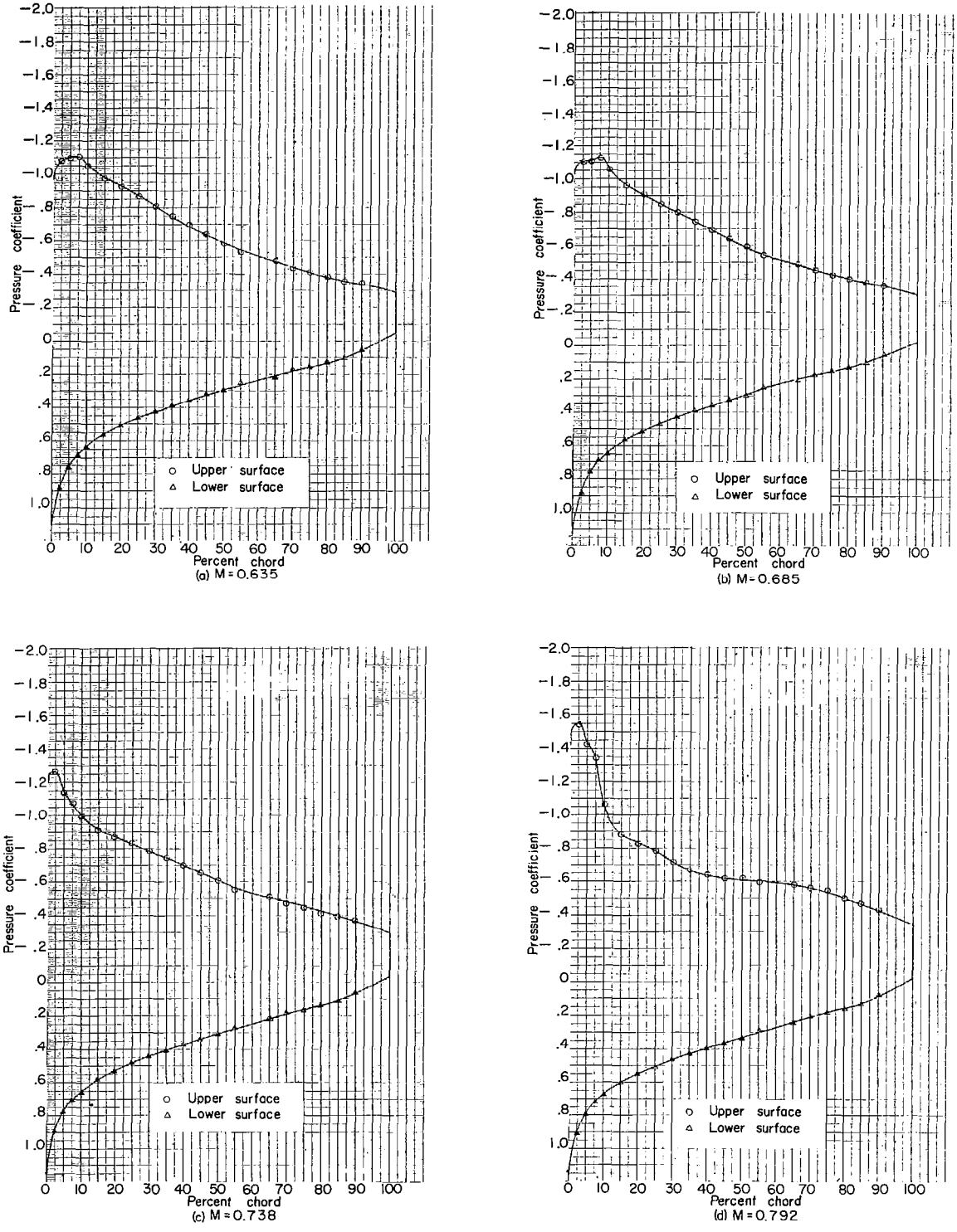
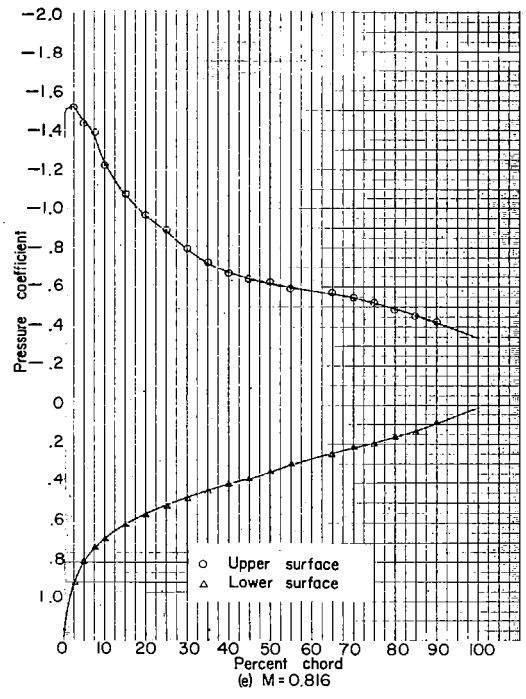
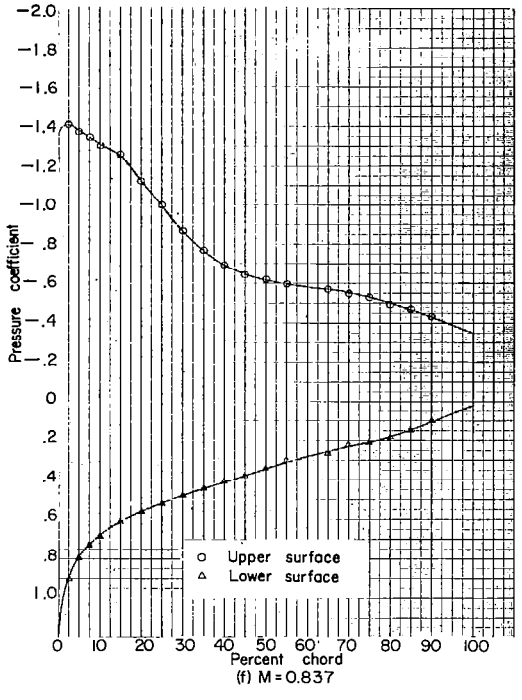


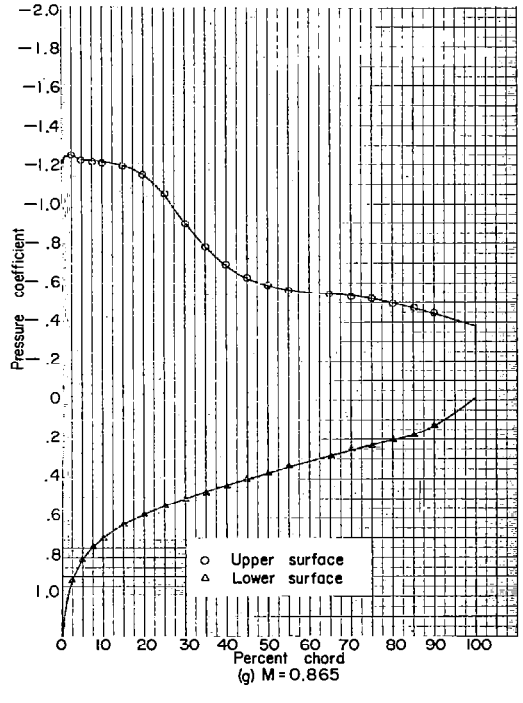
Figure 56.- Pressure distributions over NACA 16-506 airfoil sections. $\alpha = 12^{\circ}$.



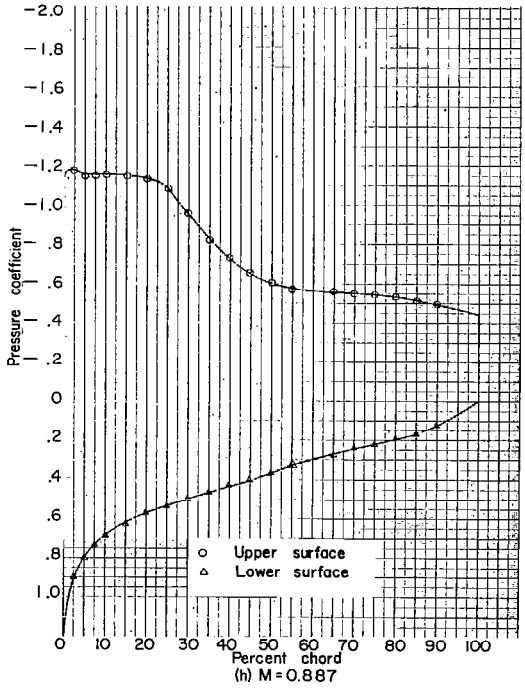
(e) M = 0.816



(f) M = 0.837

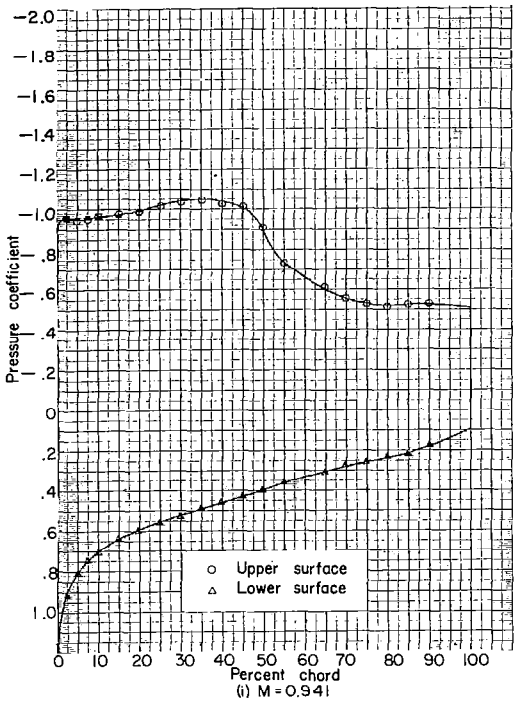


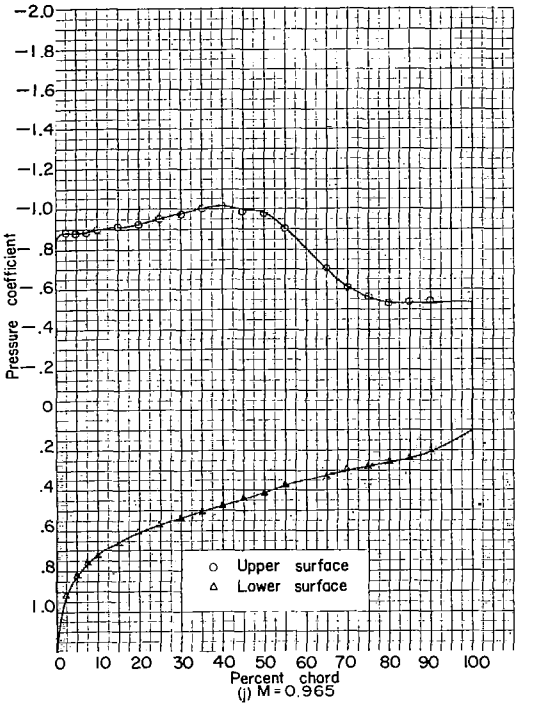
(g) M = 0.865

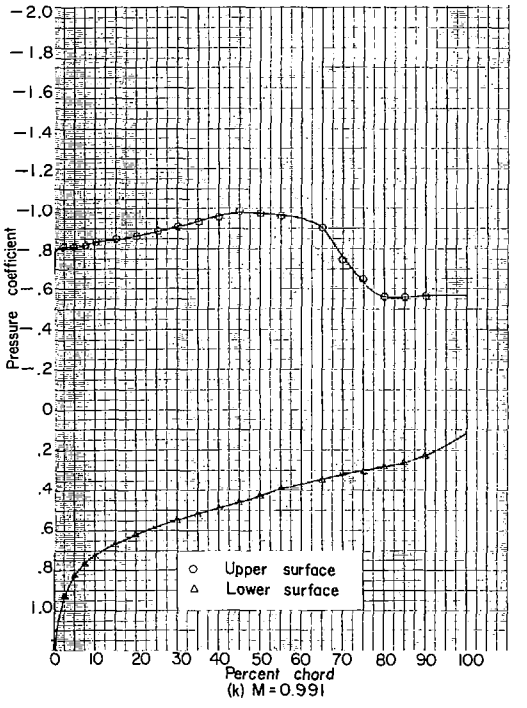


(h) M = 0.887

Figure 56.- Continued. NACA 16-506; $\alpha = 12^{\circ}$.







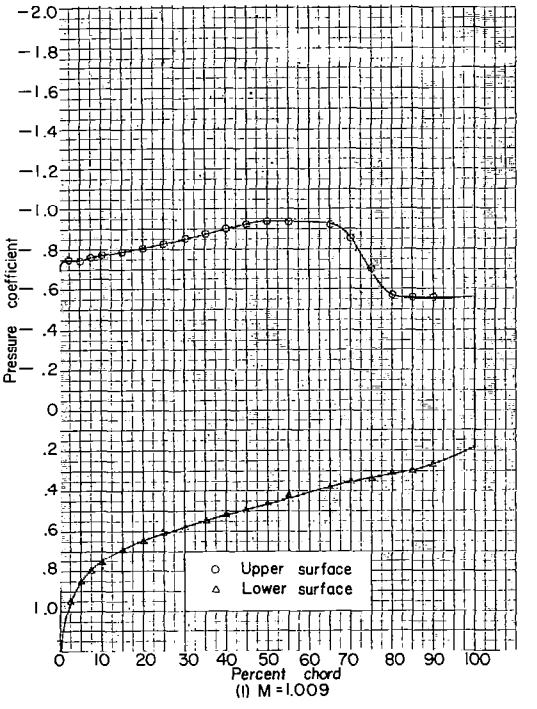
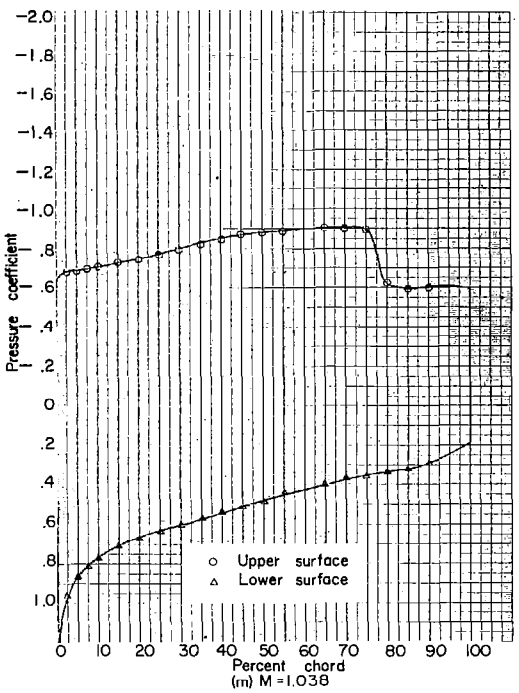
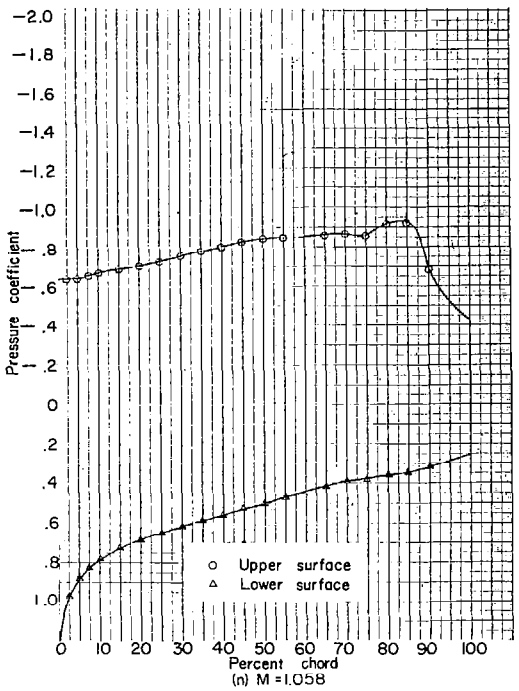
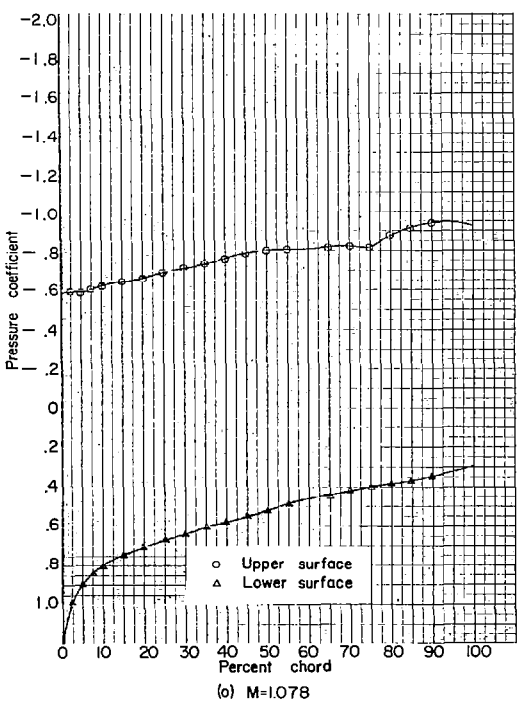


Figure 56.- Continued. NACA 16-506; $\alpha = 12^{\circ}$.







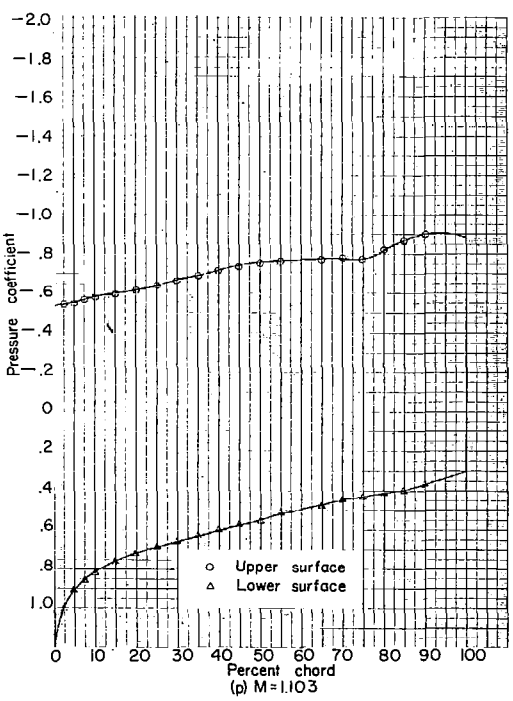


Figure 56.- Concluded. NACA 16-506; $\alpha = 12^{\circ}$.

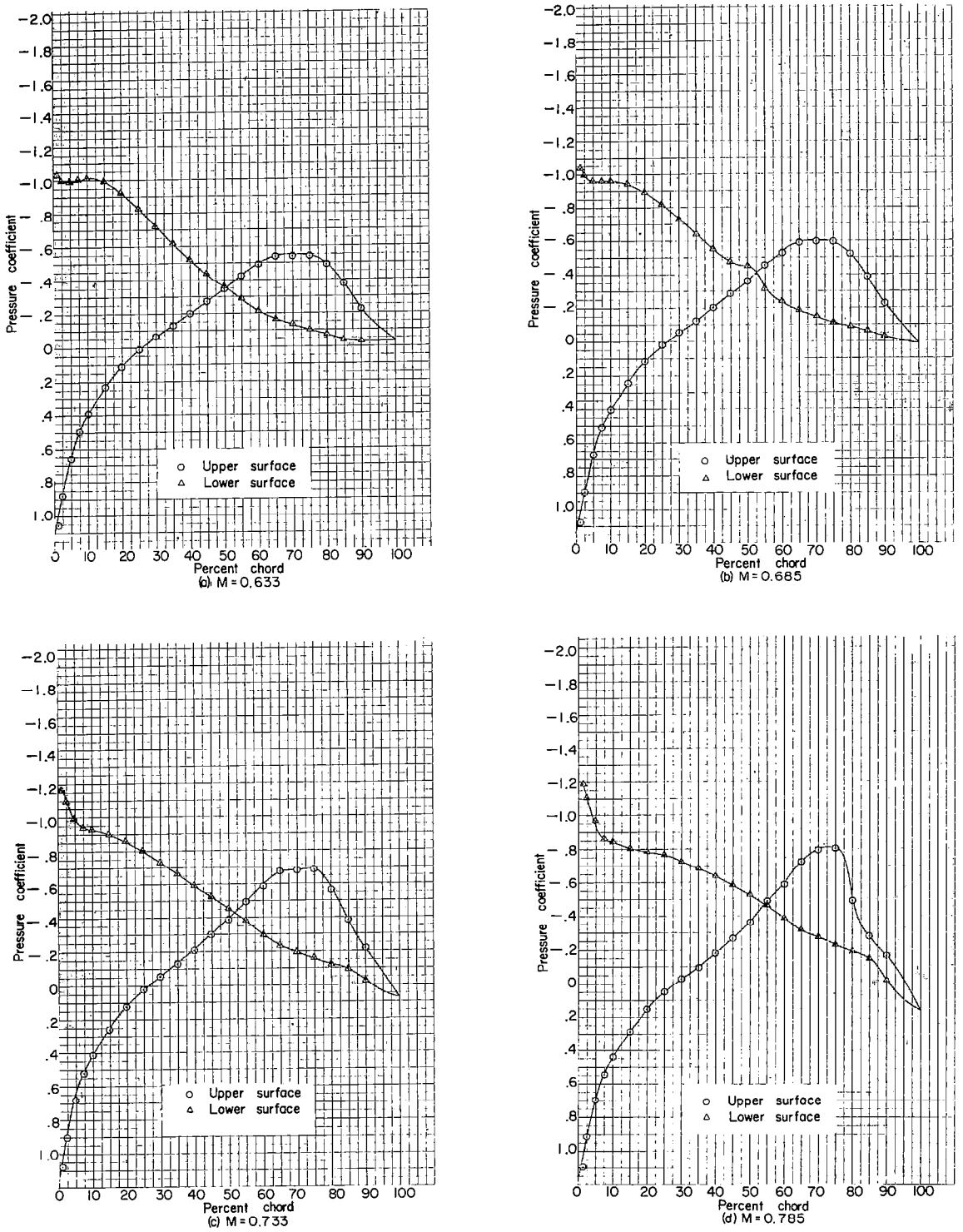
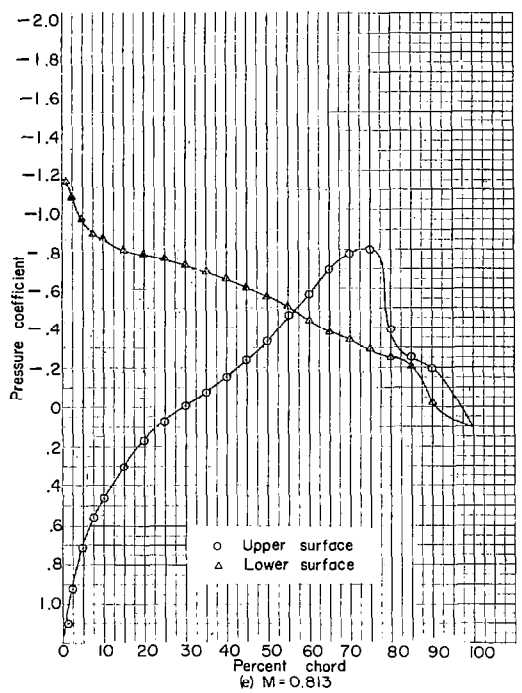
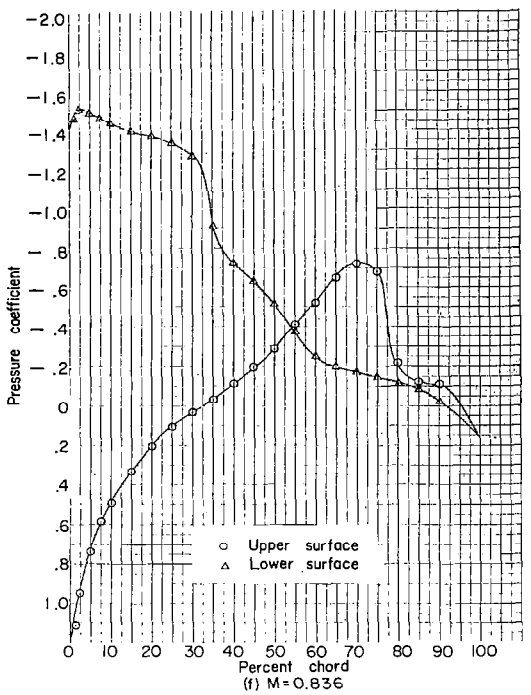
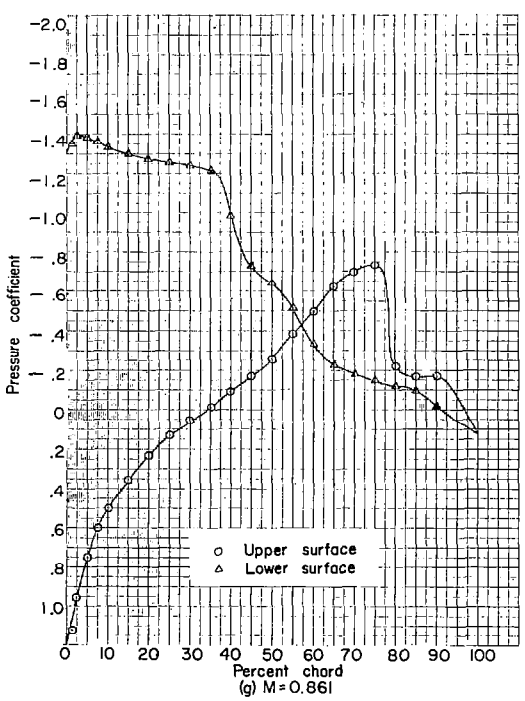


Figure 57.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = -10^{\circ}$.







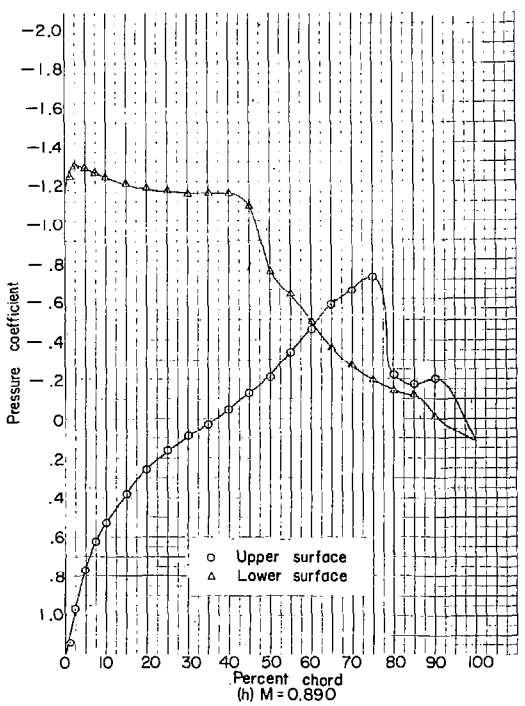
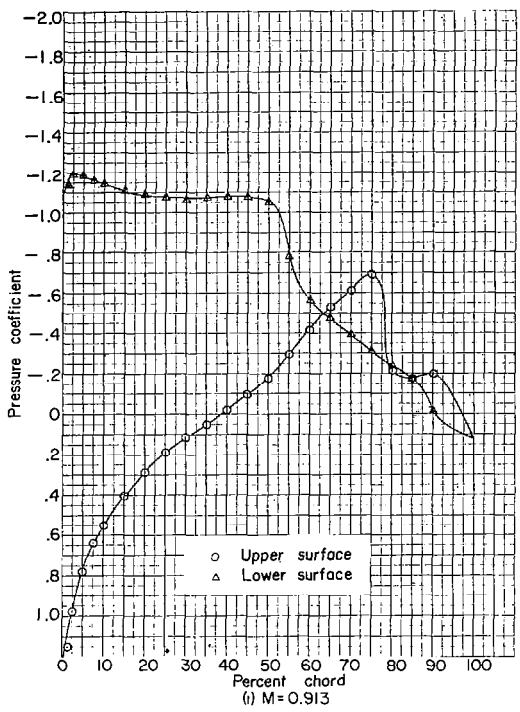
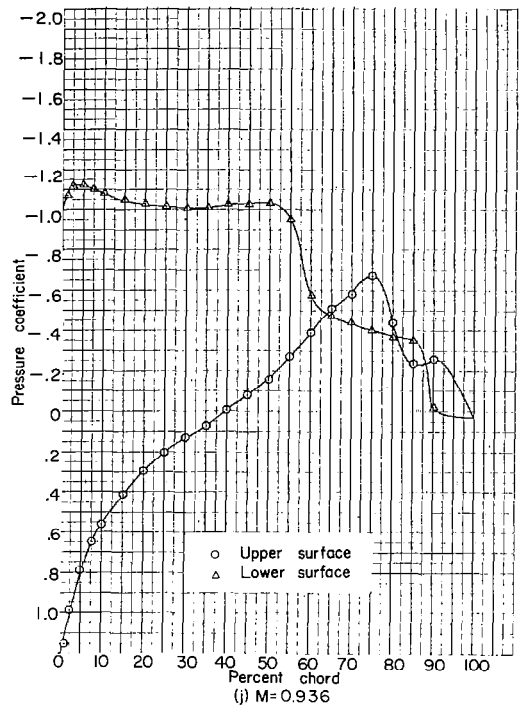
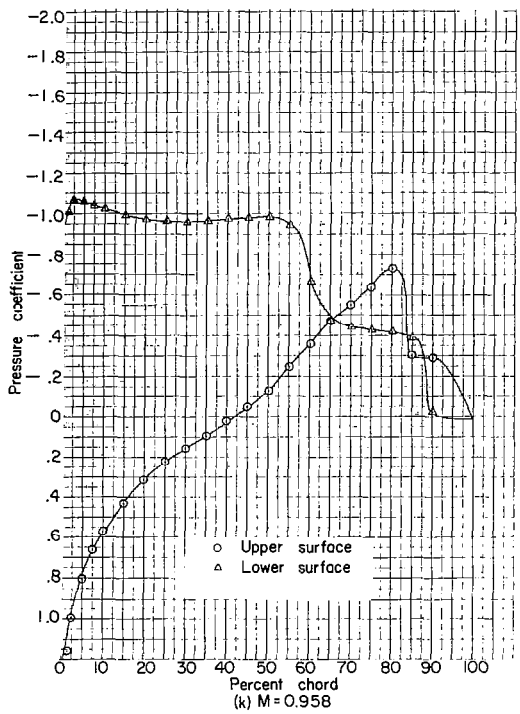


Figure 57.- Continued. NACA 16-512; $\alpha = -10^{\circ}$.







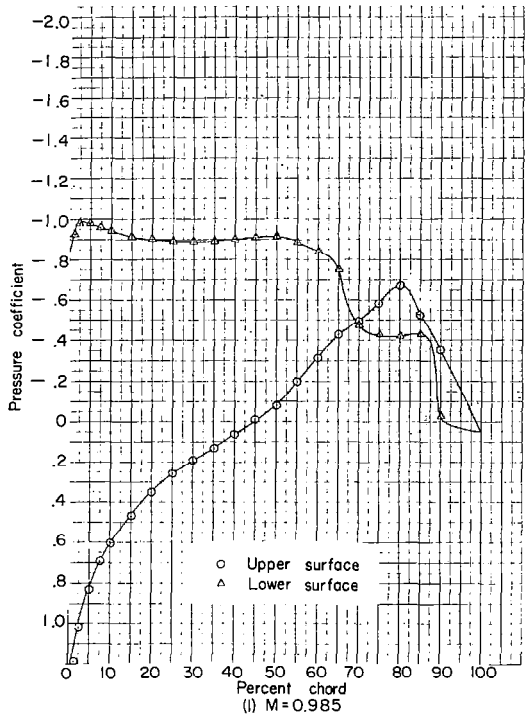
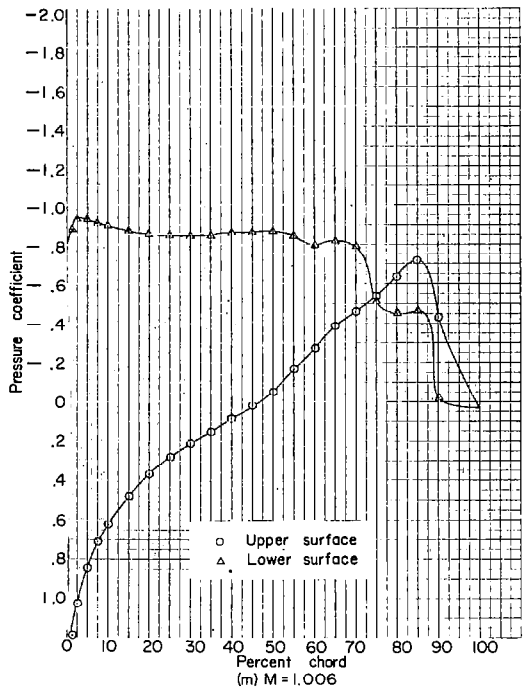
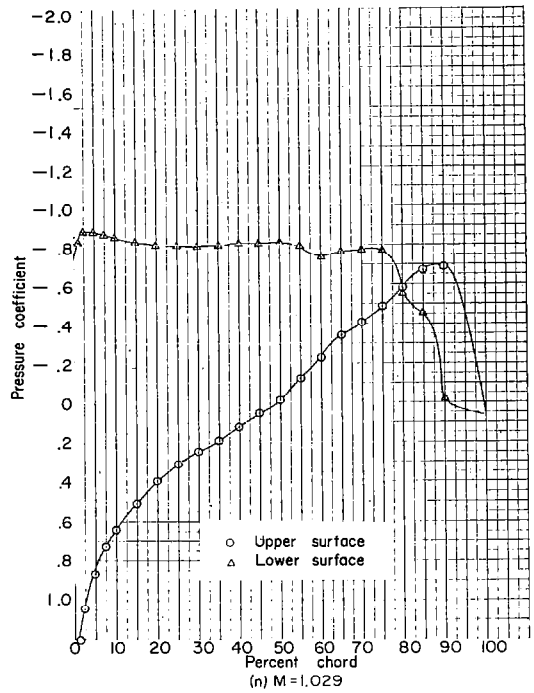
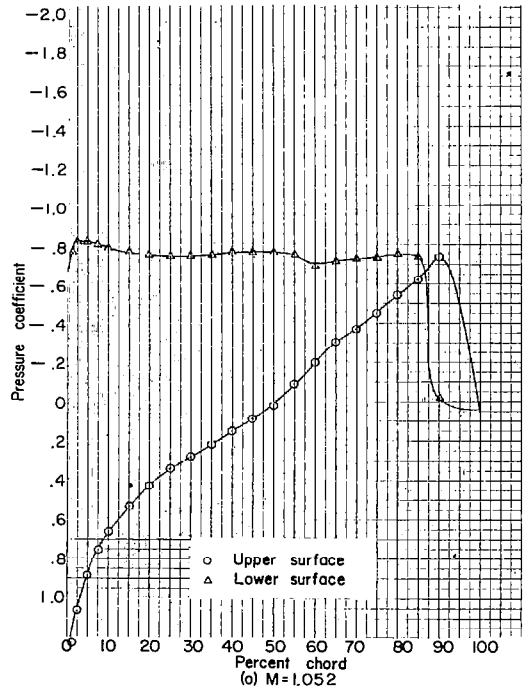


Figure 57.- Continued. NACA 16-512; $\alpha = -10^{\circ}$.







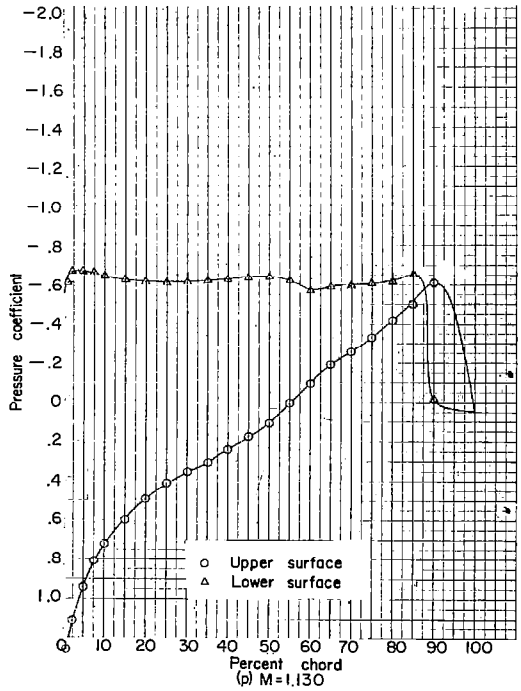


Figure 57.- Concluded. NACA 16-512; $\alpha = -10^{\circ}$.

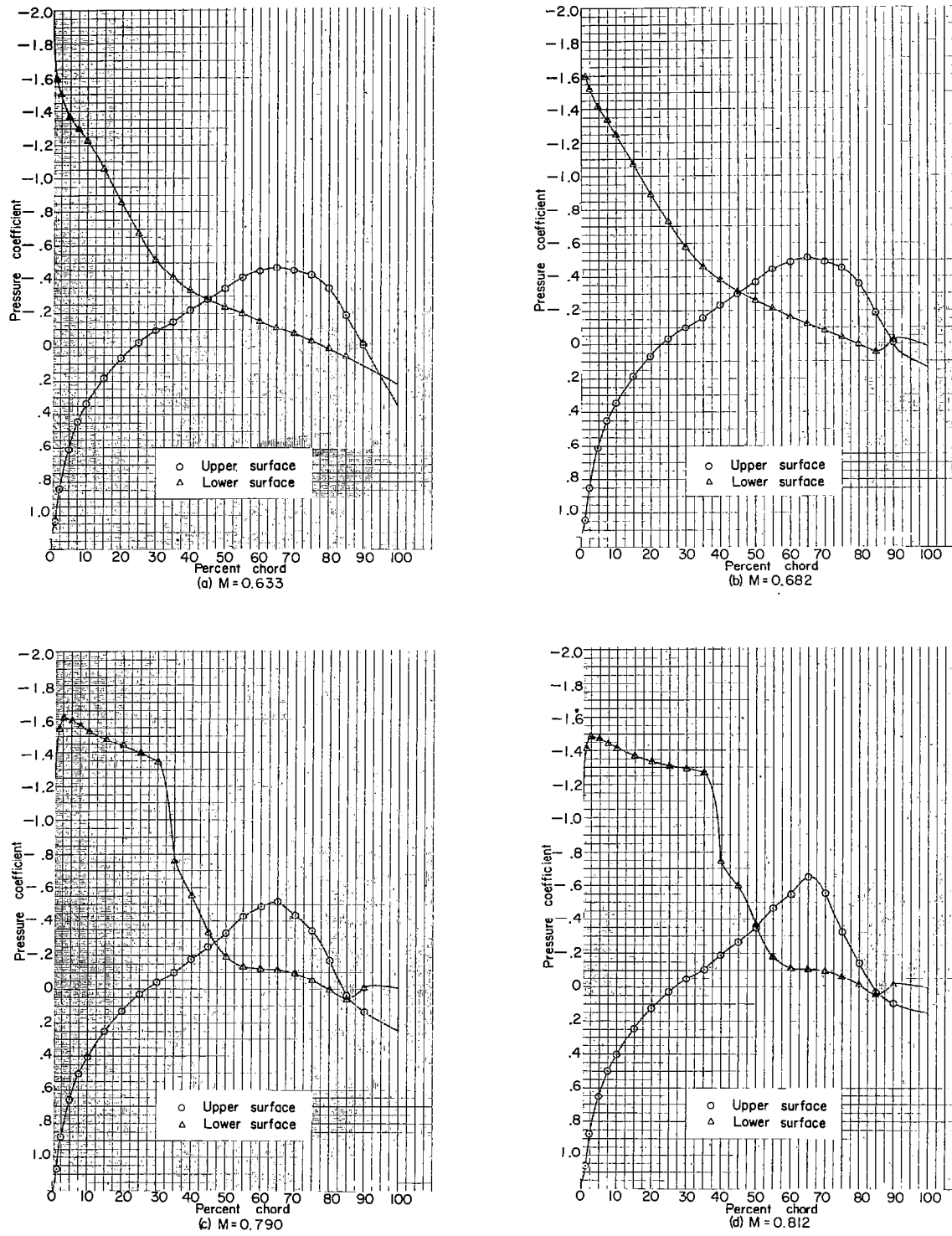
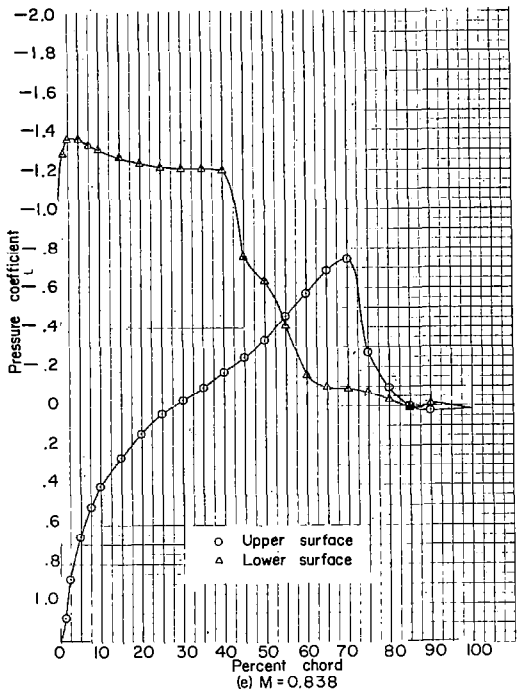
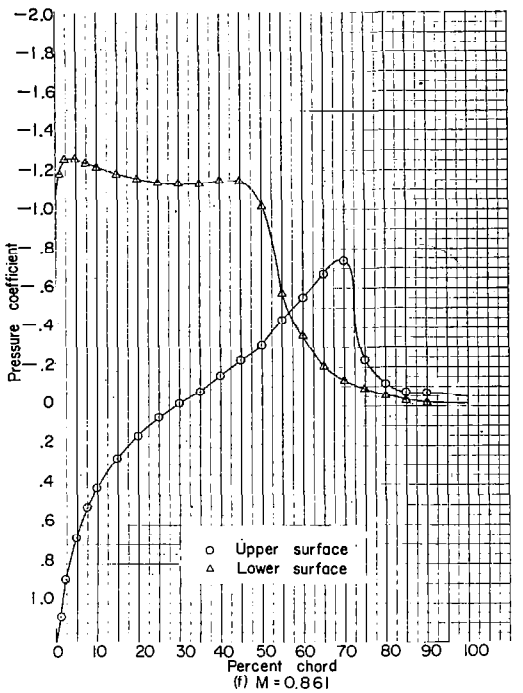
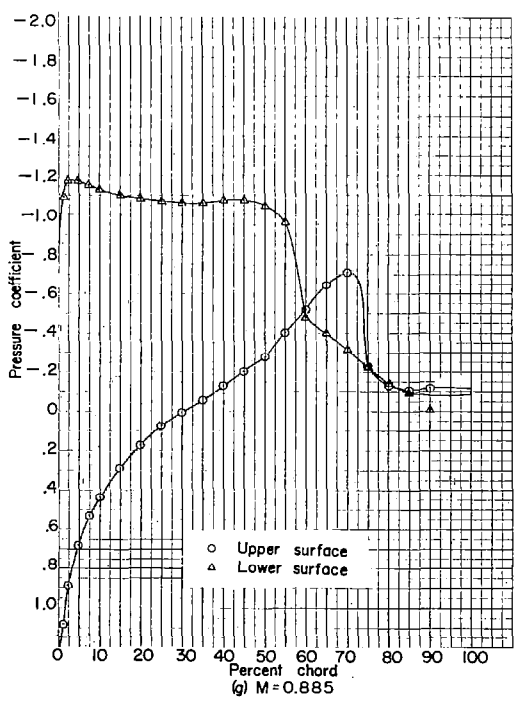


Figure 58.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = -8^{\circ}$.







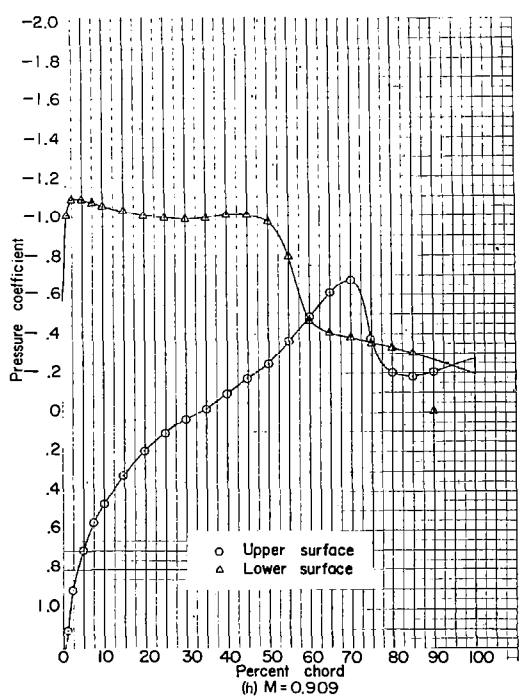


Figure 58.- Continued. NACA 16-512; $\alpha = -8^{\circ}$.

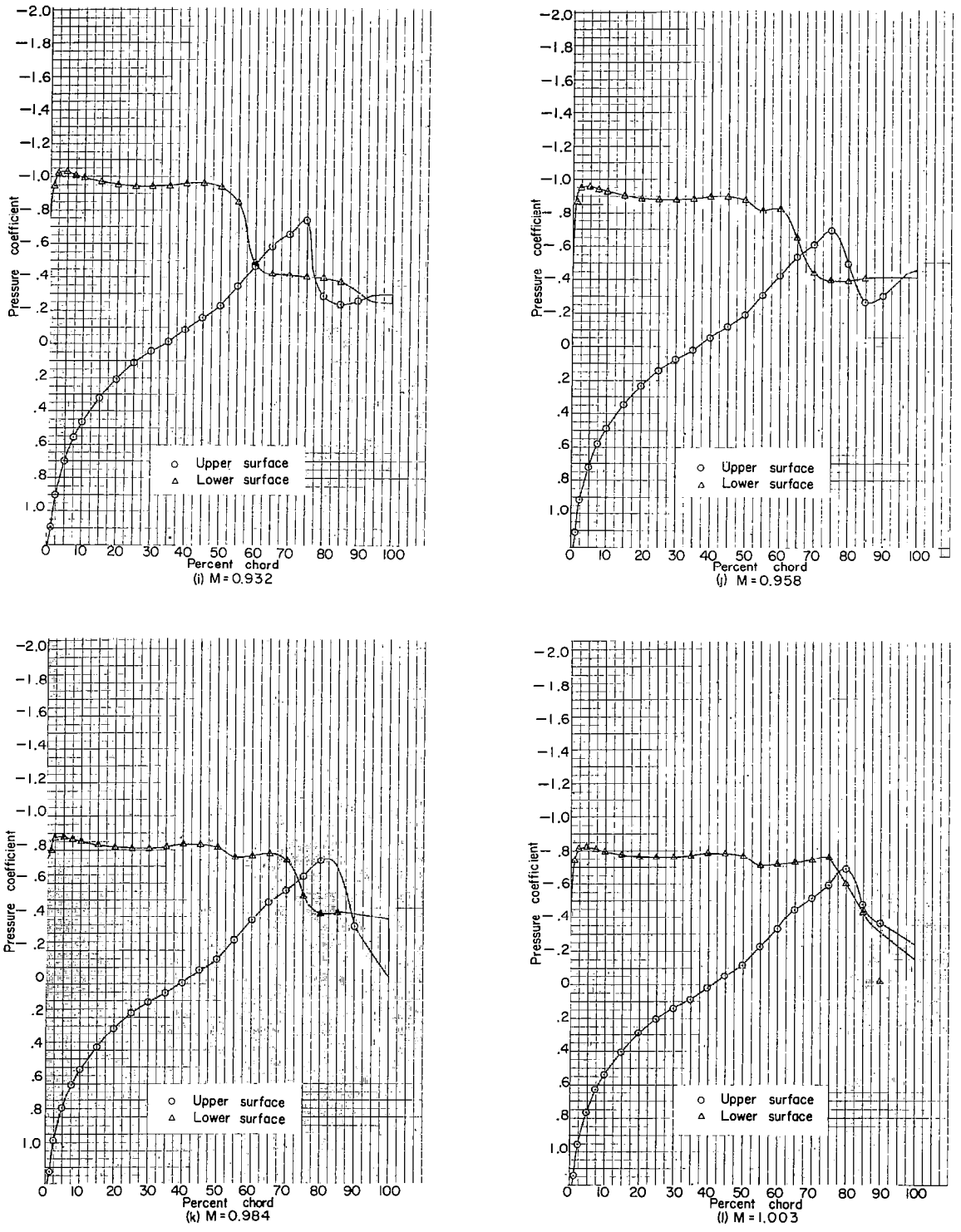
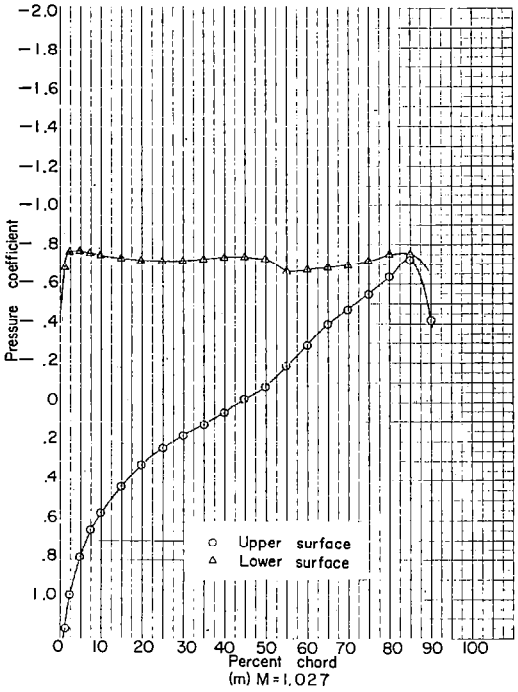
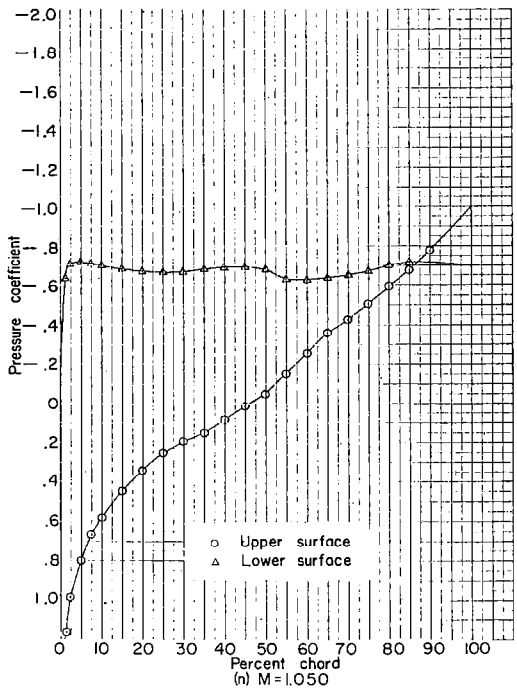
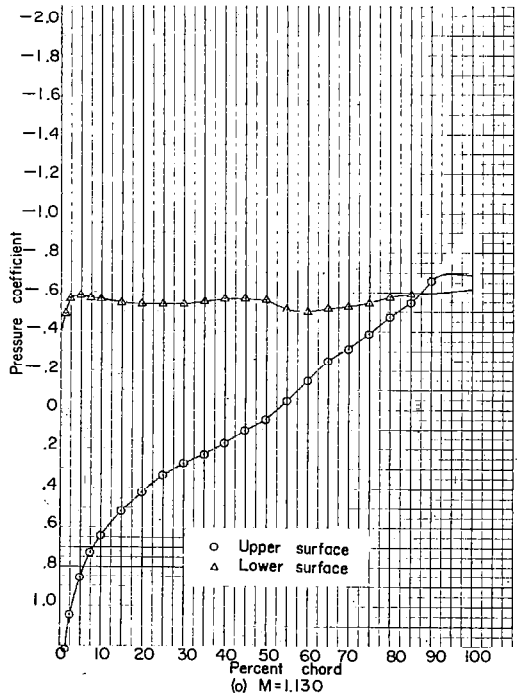


Figure 58.- Continued. NACA 16-512; $\alpha = -8^{\circ}$.







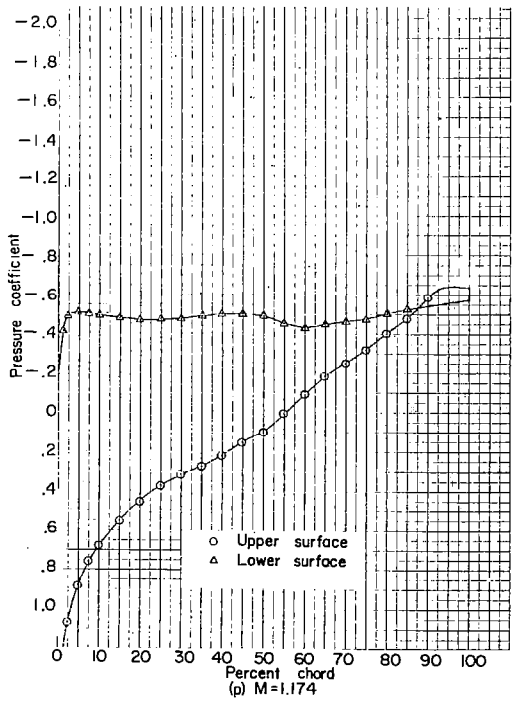


Figure 58.- Concluded. NACA 16-512; $\alpha = -8^{\circ}$.

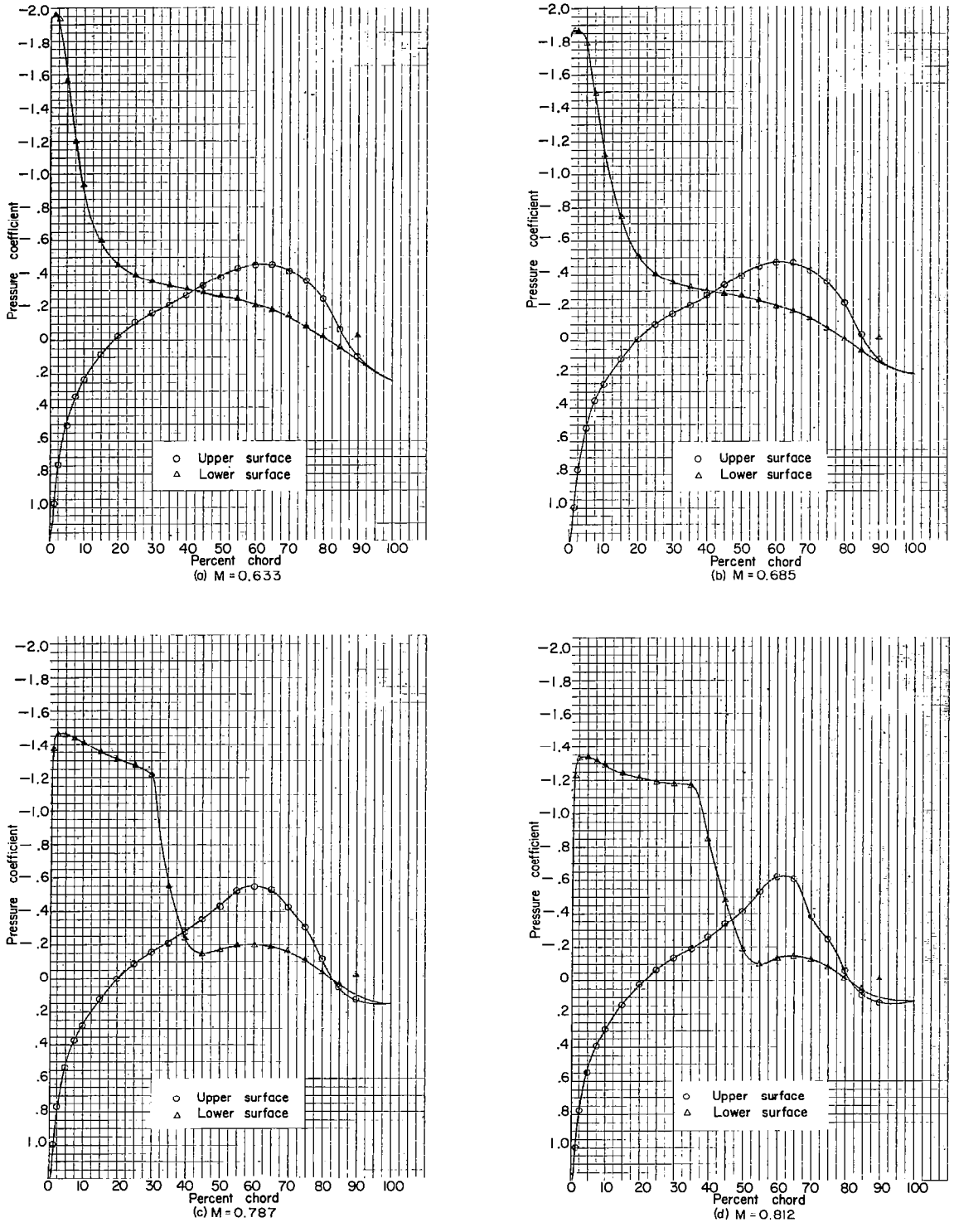
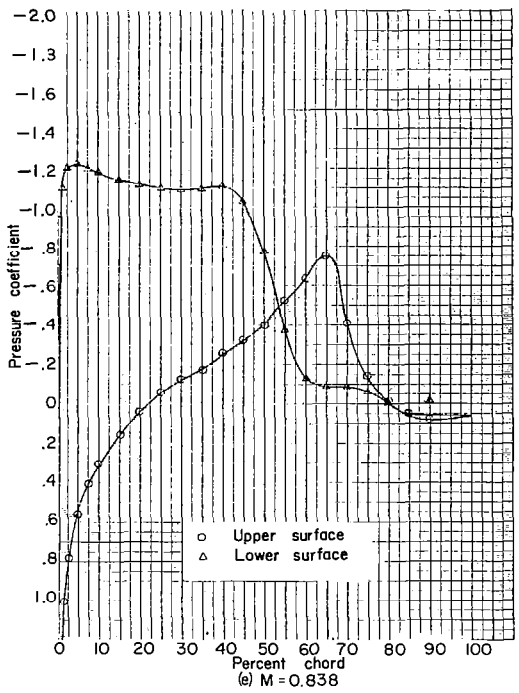
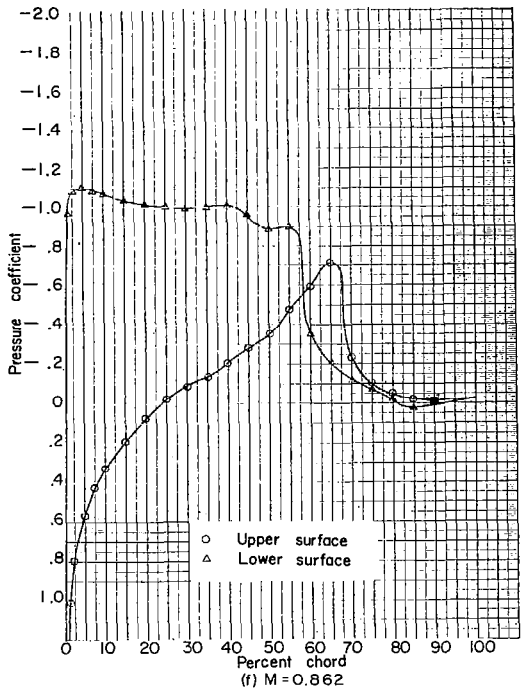
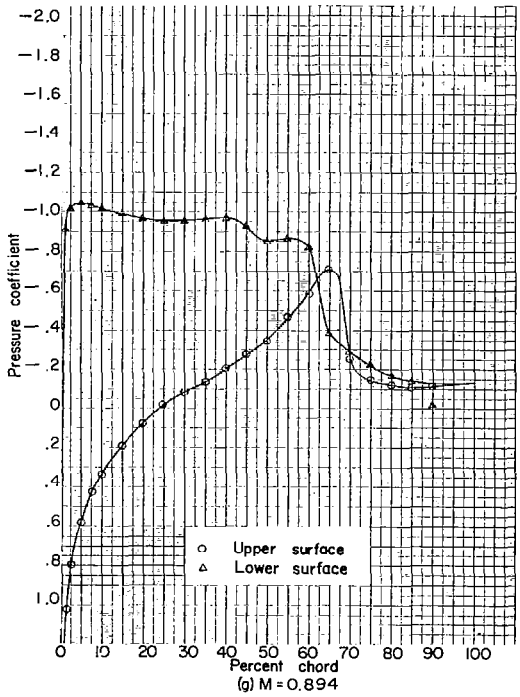


Figure 59.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = -6^{\circ}$.







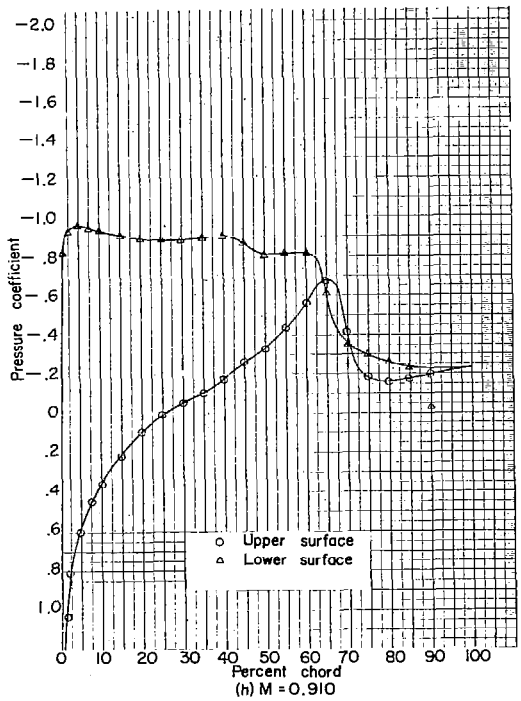
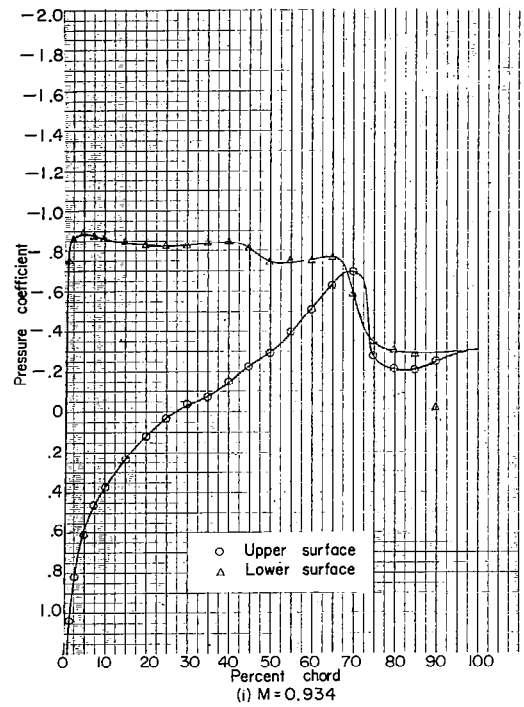
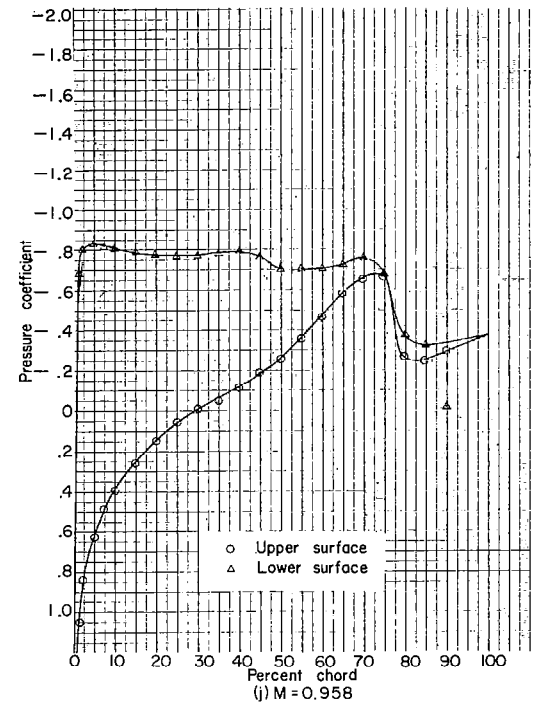
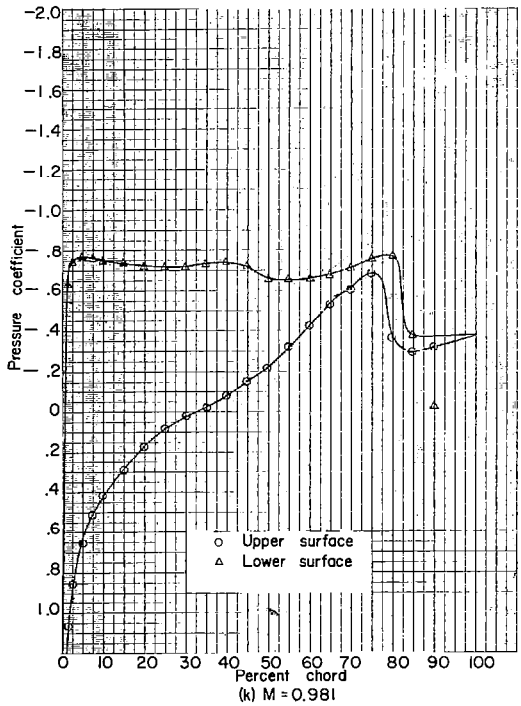


Figure 59.- Continued. NACA 16-512; $\alpha = -6^{\circ}$.







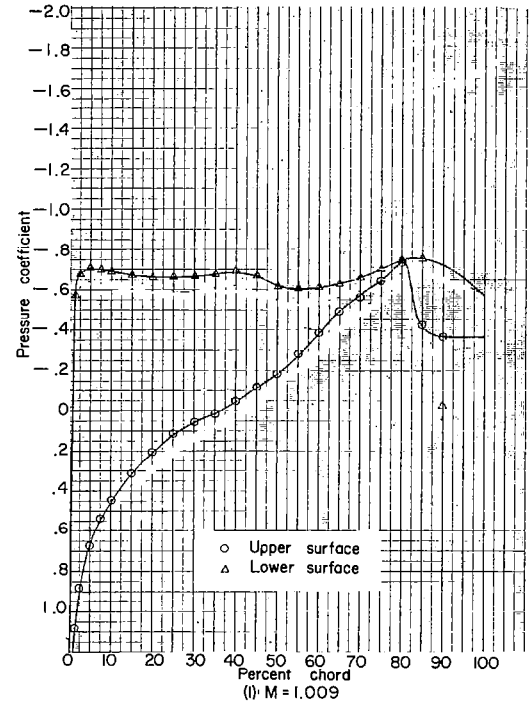
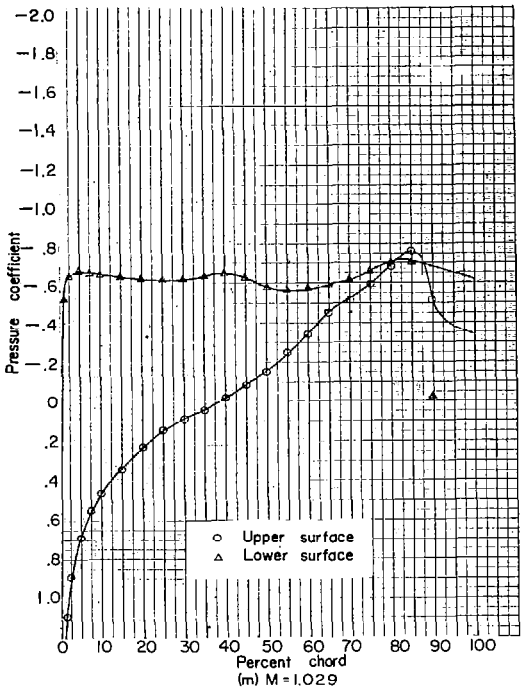
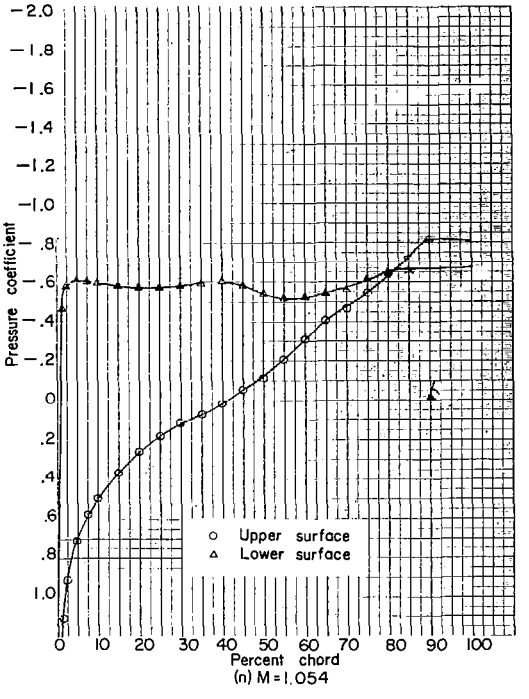
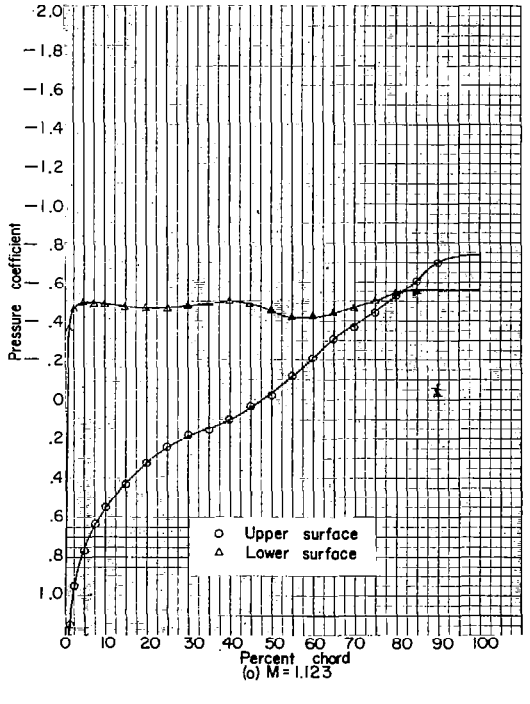


Figure 59.- Continued. NACA 16-512; $\alpha = -6^{\circ}$.







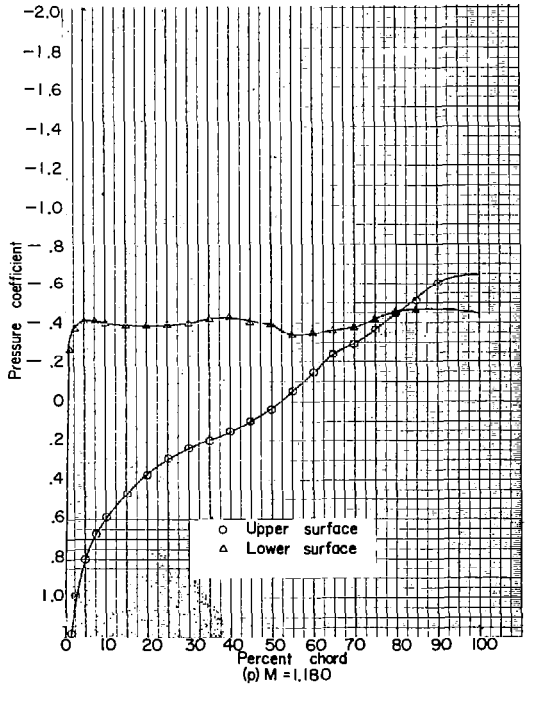


Figure 59.- Concluded. NACA 16-512; $\alpha = -6^{\circ}$.

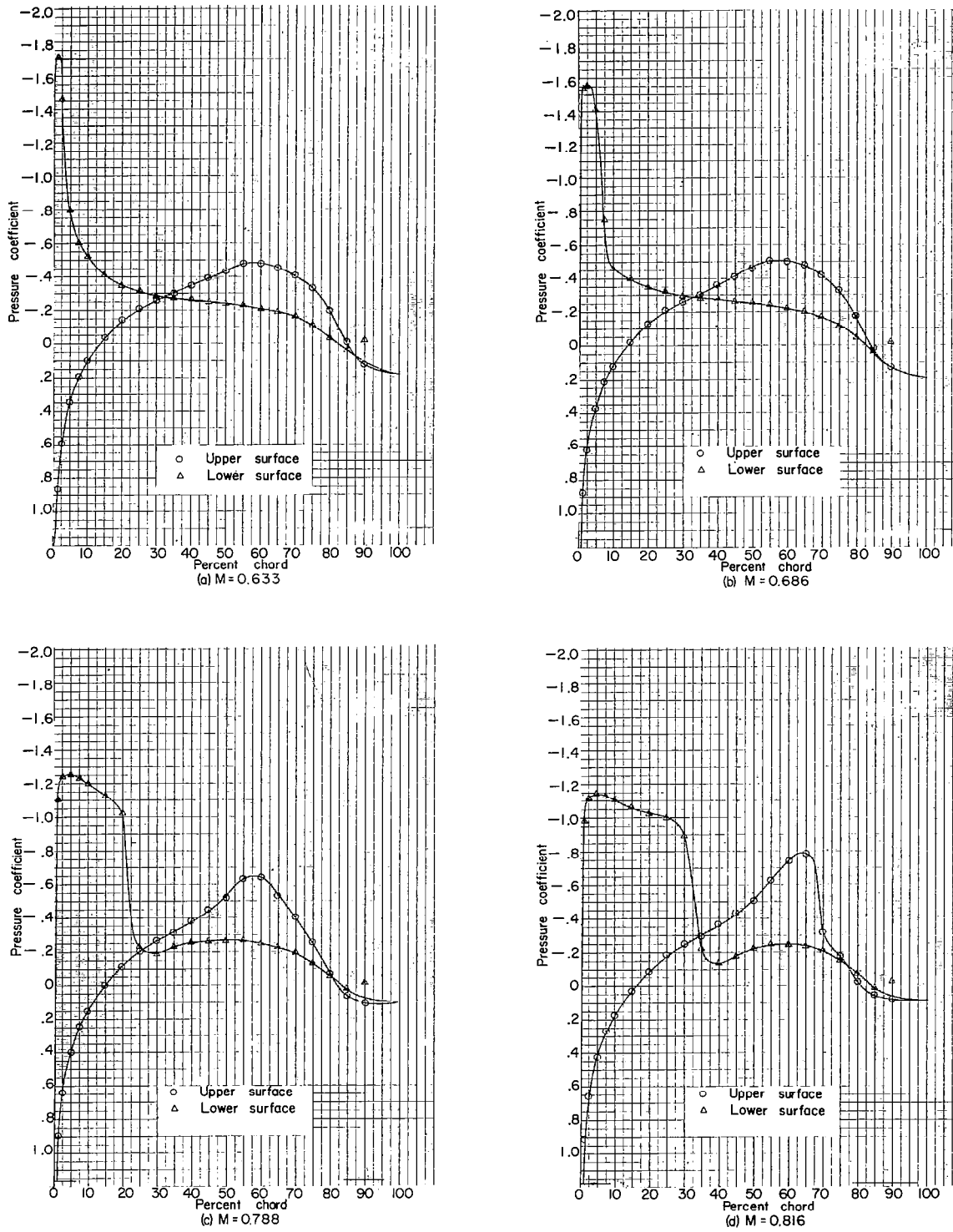
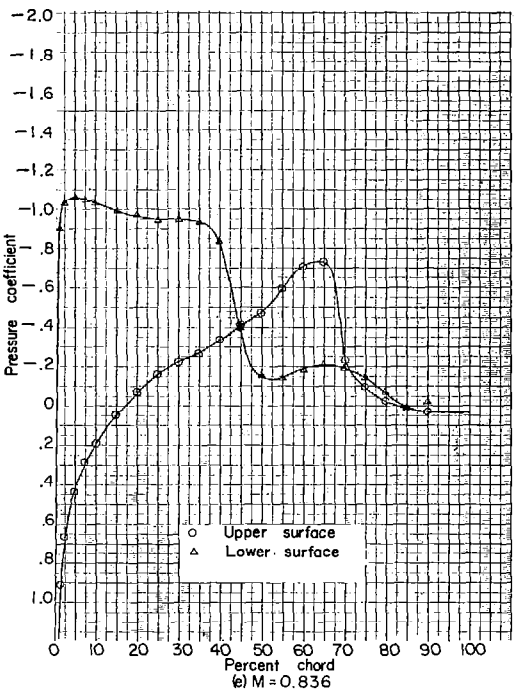
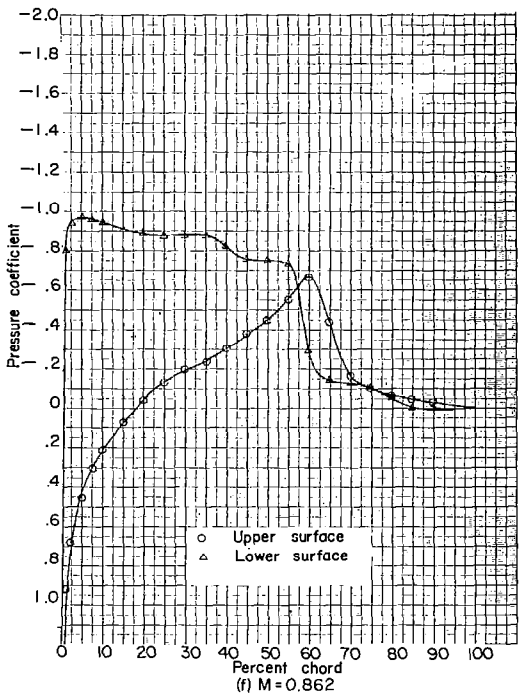
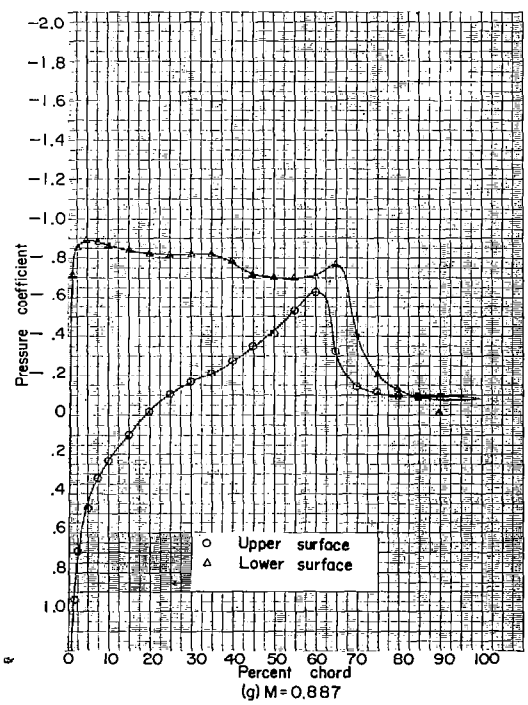


Figure 60.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = -4^{\circ}$.







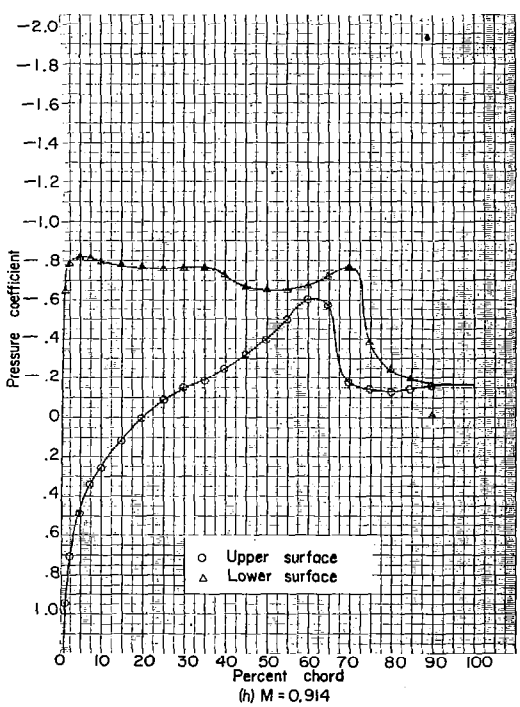
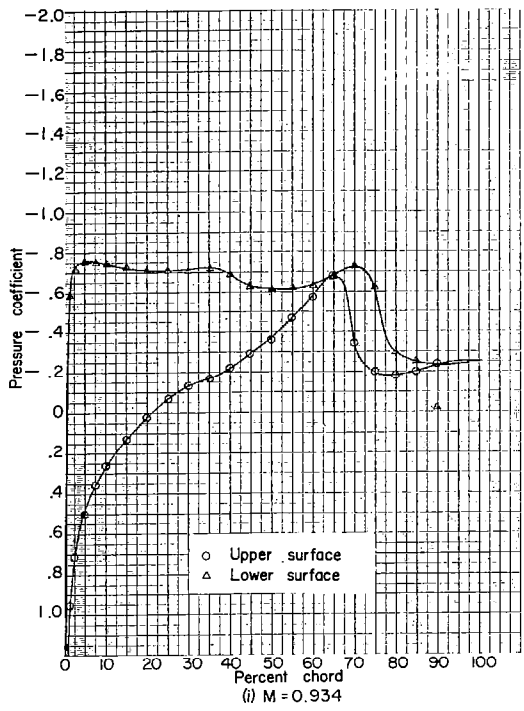
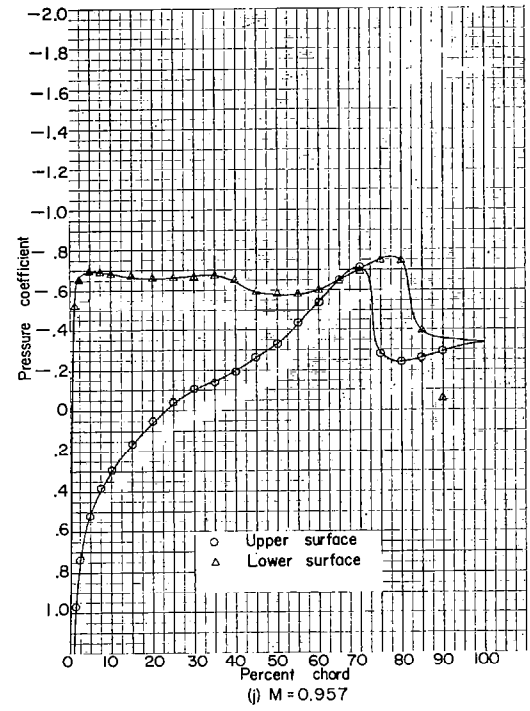
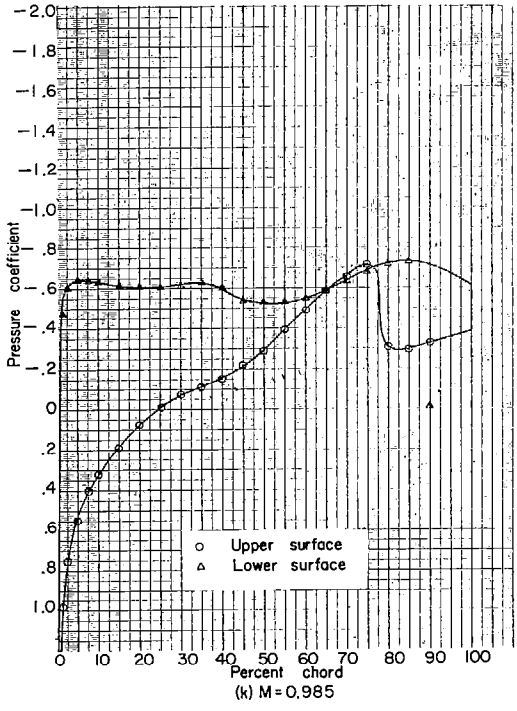


Figure 60.- Continued. NACA 16-512; $\alpha = -4^{\circ}$.







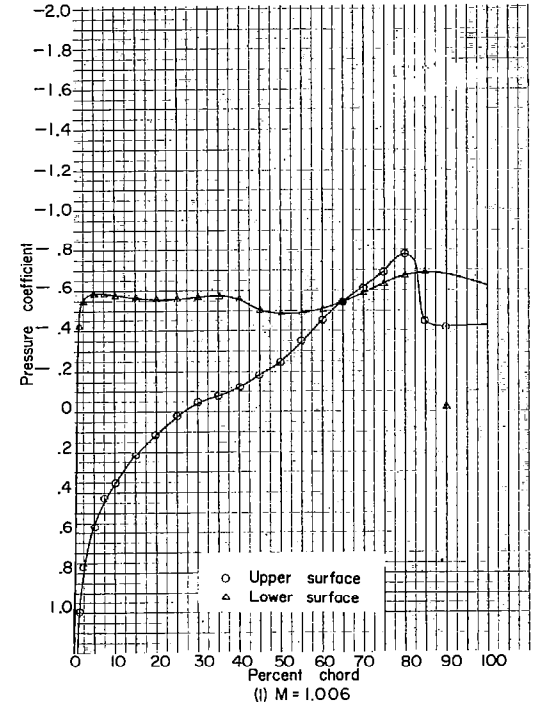
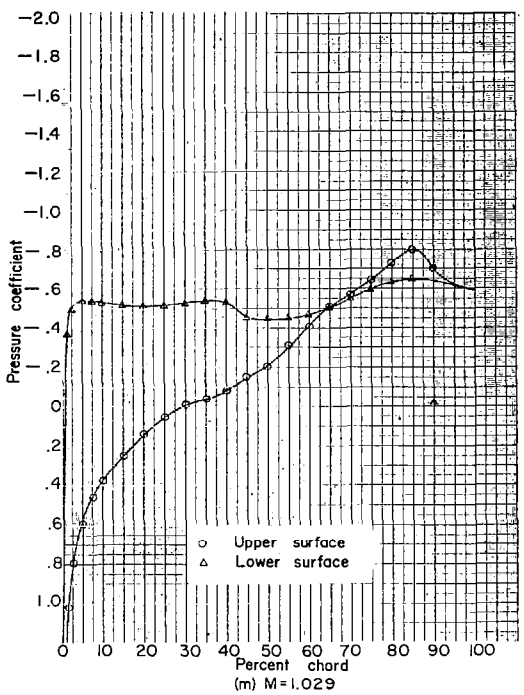
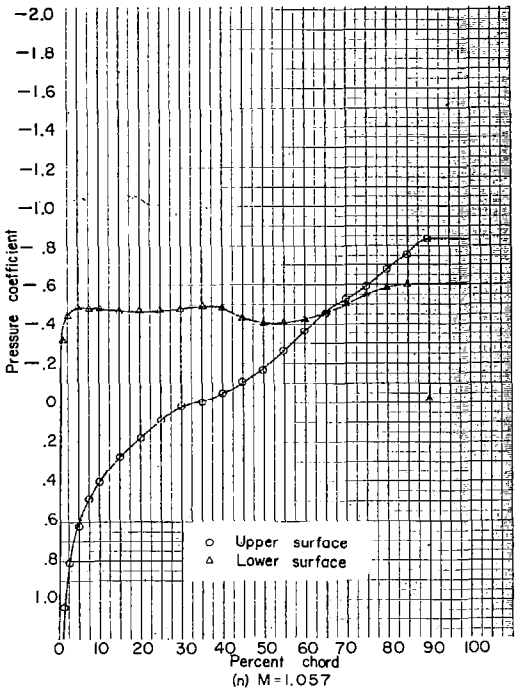


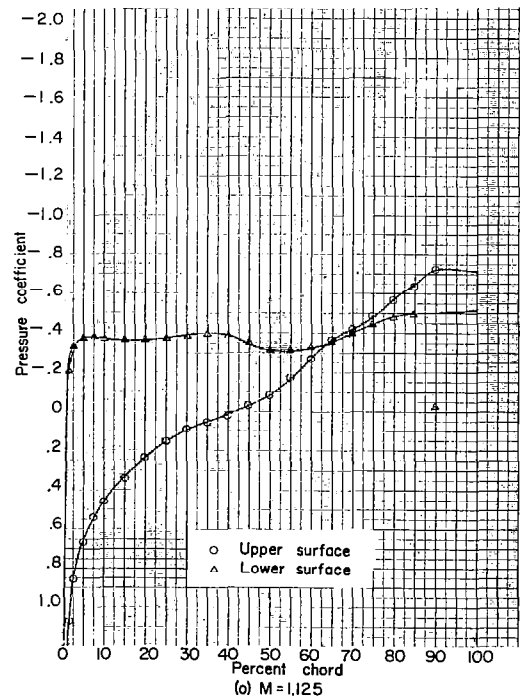
Figure 60.- Continued. NACA 16-512; $\alpha = -4^{\circ}$.



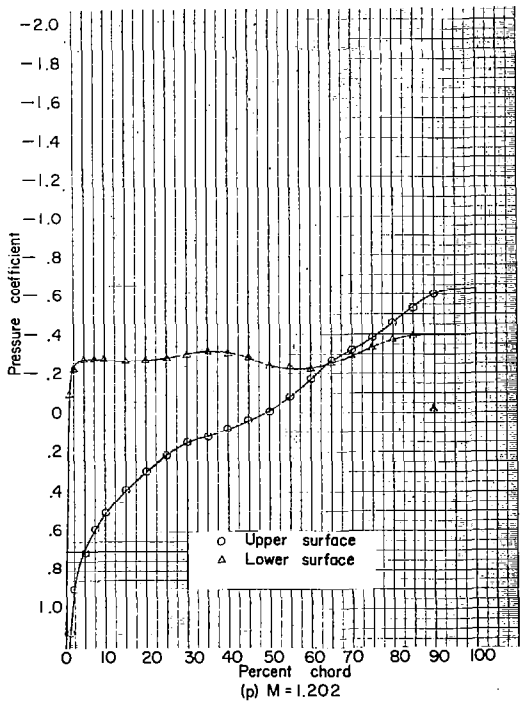
(m) M = 1.029



(n) M = 1.057



(o) M = 1.125



(p) M = 1.202

Figure 60.- Concluded. NACA 16-512; $\alpha = -4^{\circ}$.

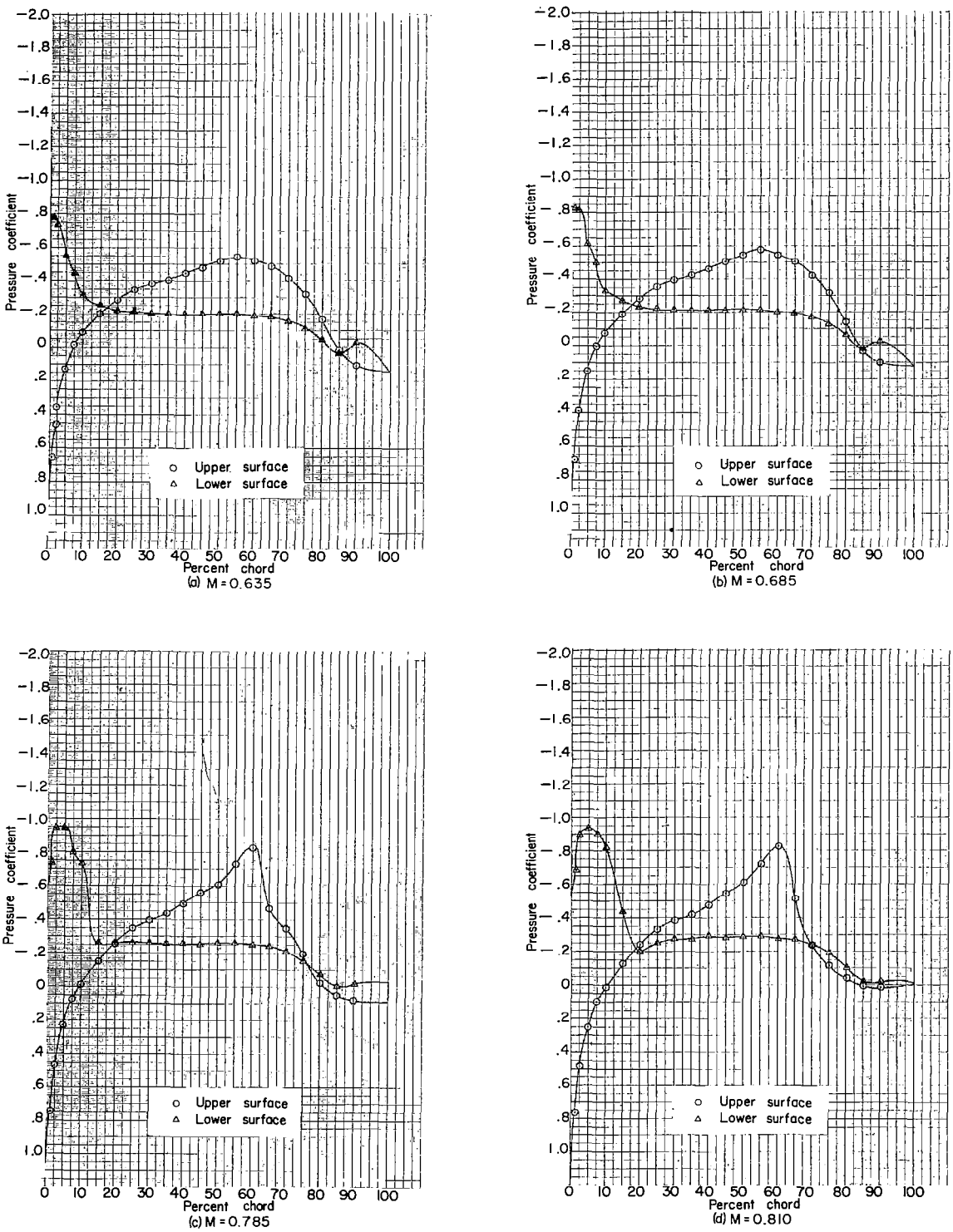
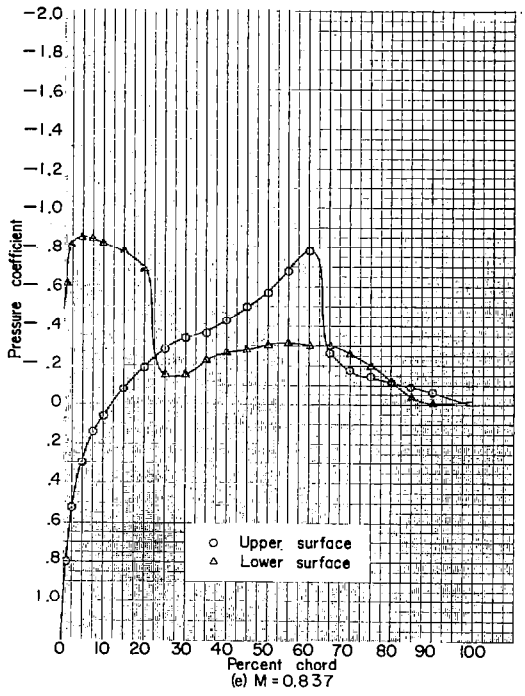
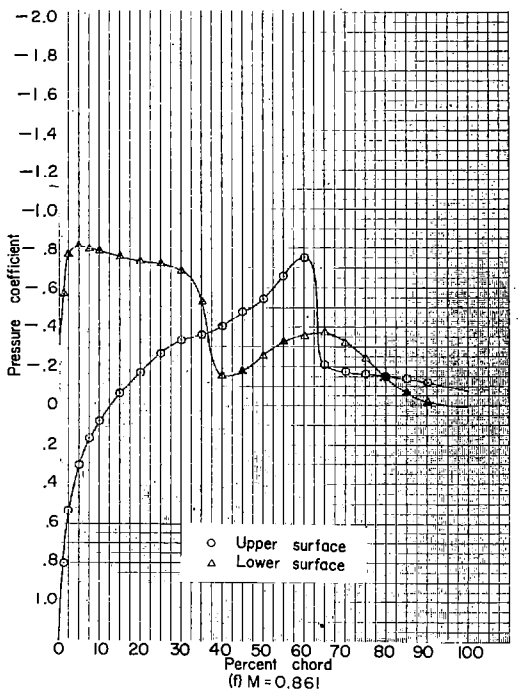
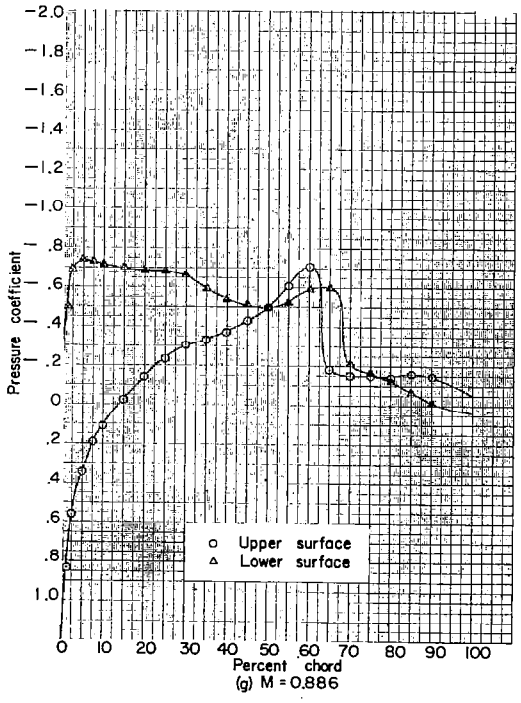


Figure 61.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = -2^{\circ}$.







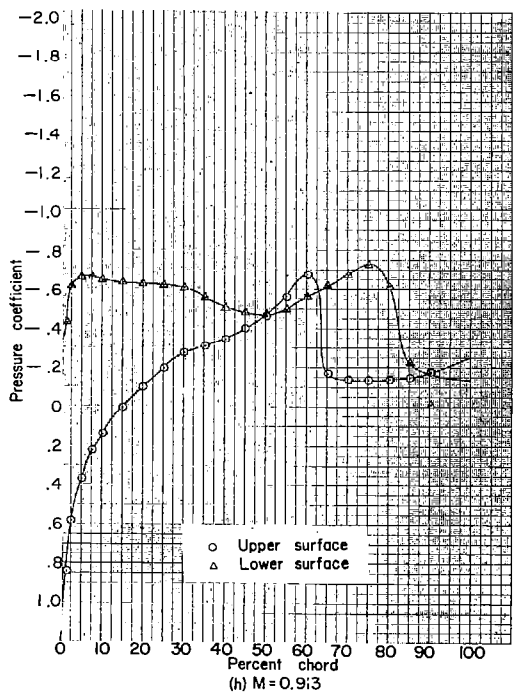
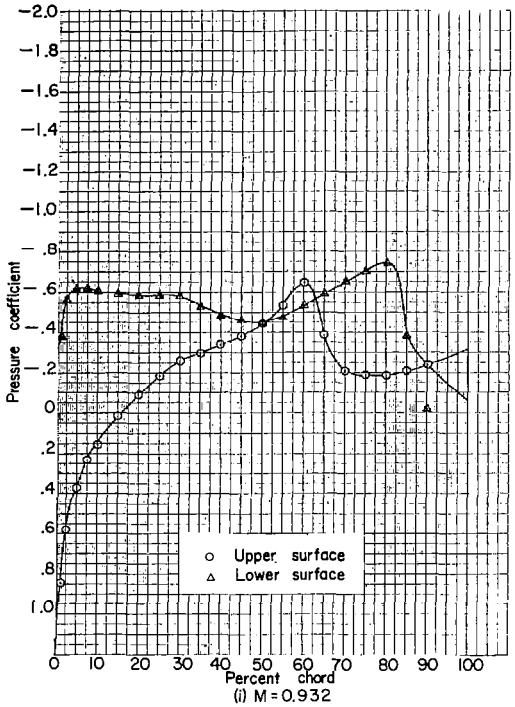
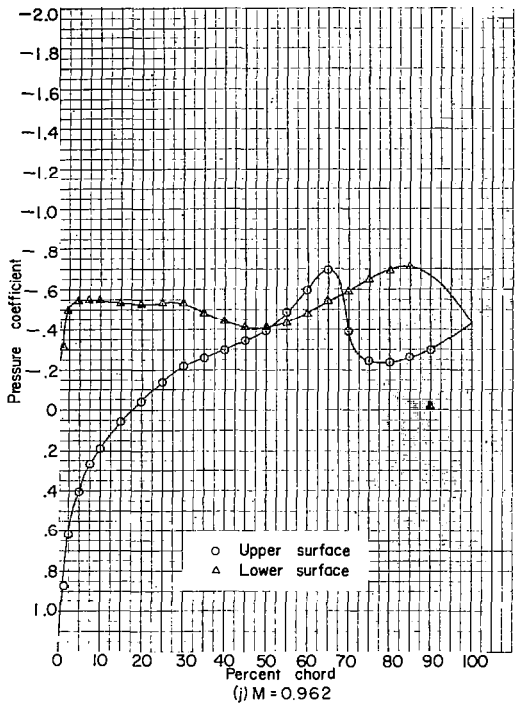
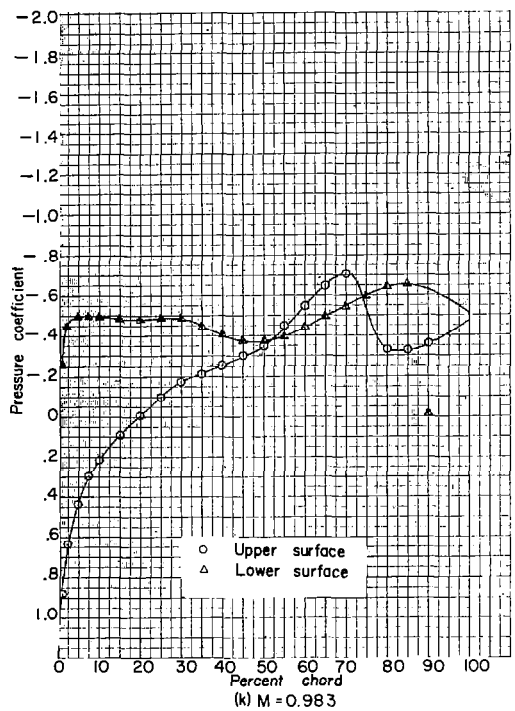


Figure 61.- Continued. NACA 16-512; $\alpha = -2^{\circ}$.







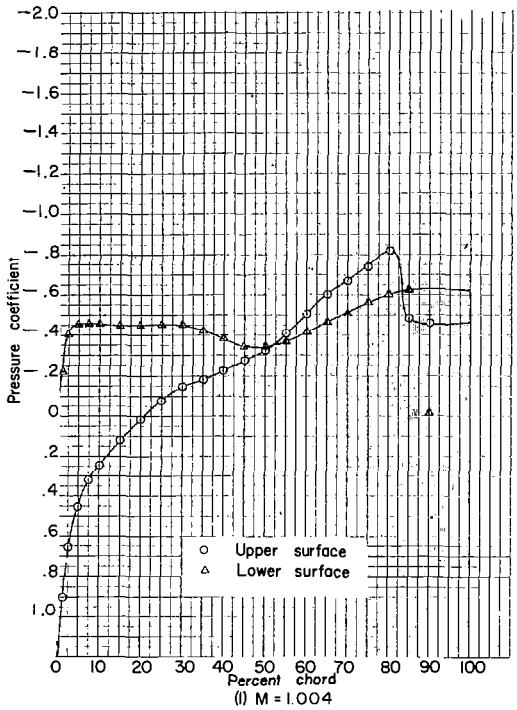
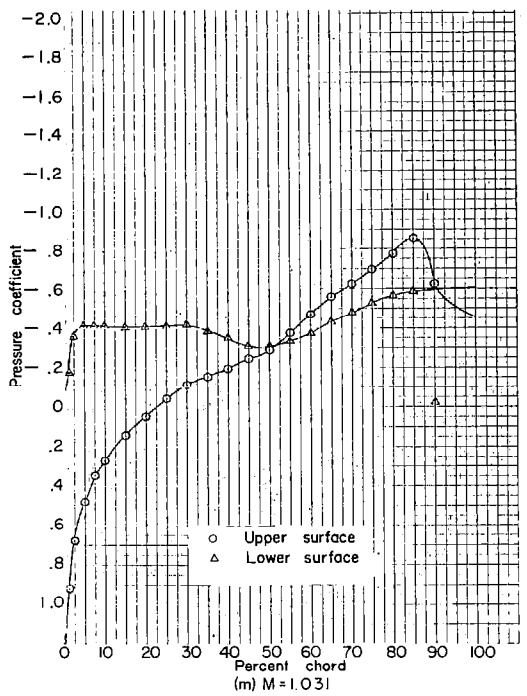
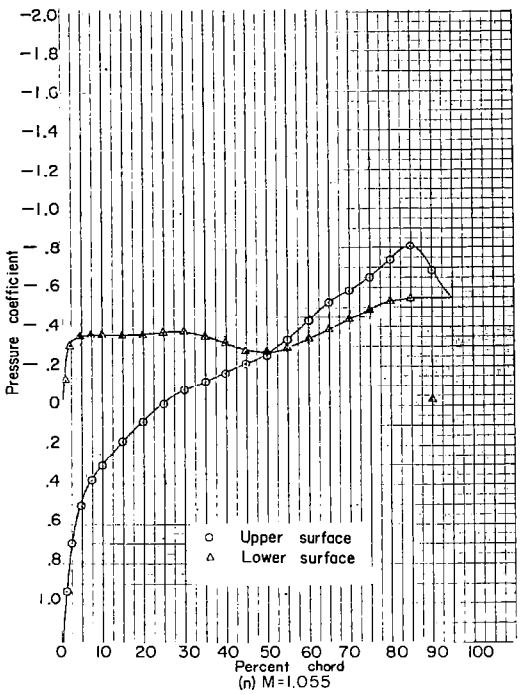
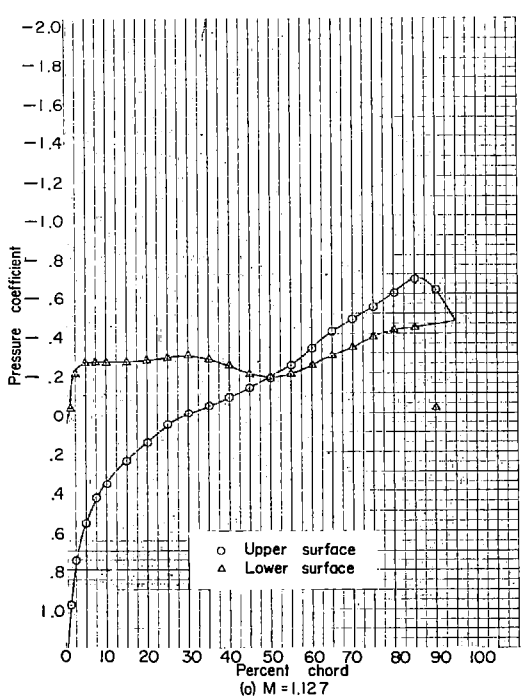


Figure 61.- Continued. NACA 16-512; $\alpha = -2^{\circ}$.







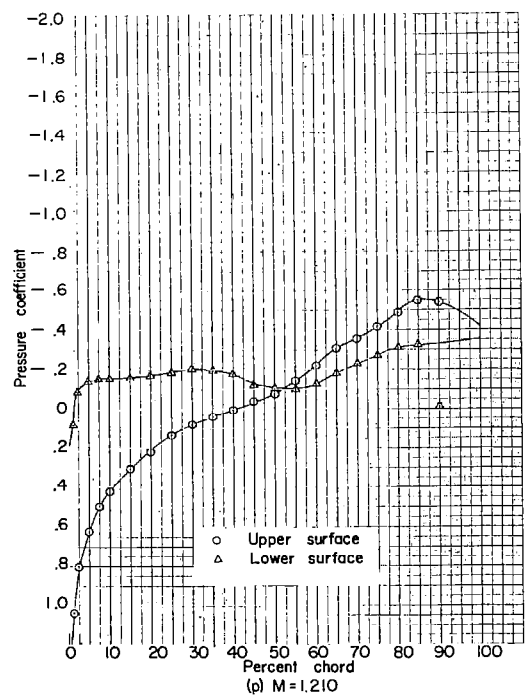


Figure 61.- Concluded. NACA 16-512; $\alpha = -2^{\circ}$.

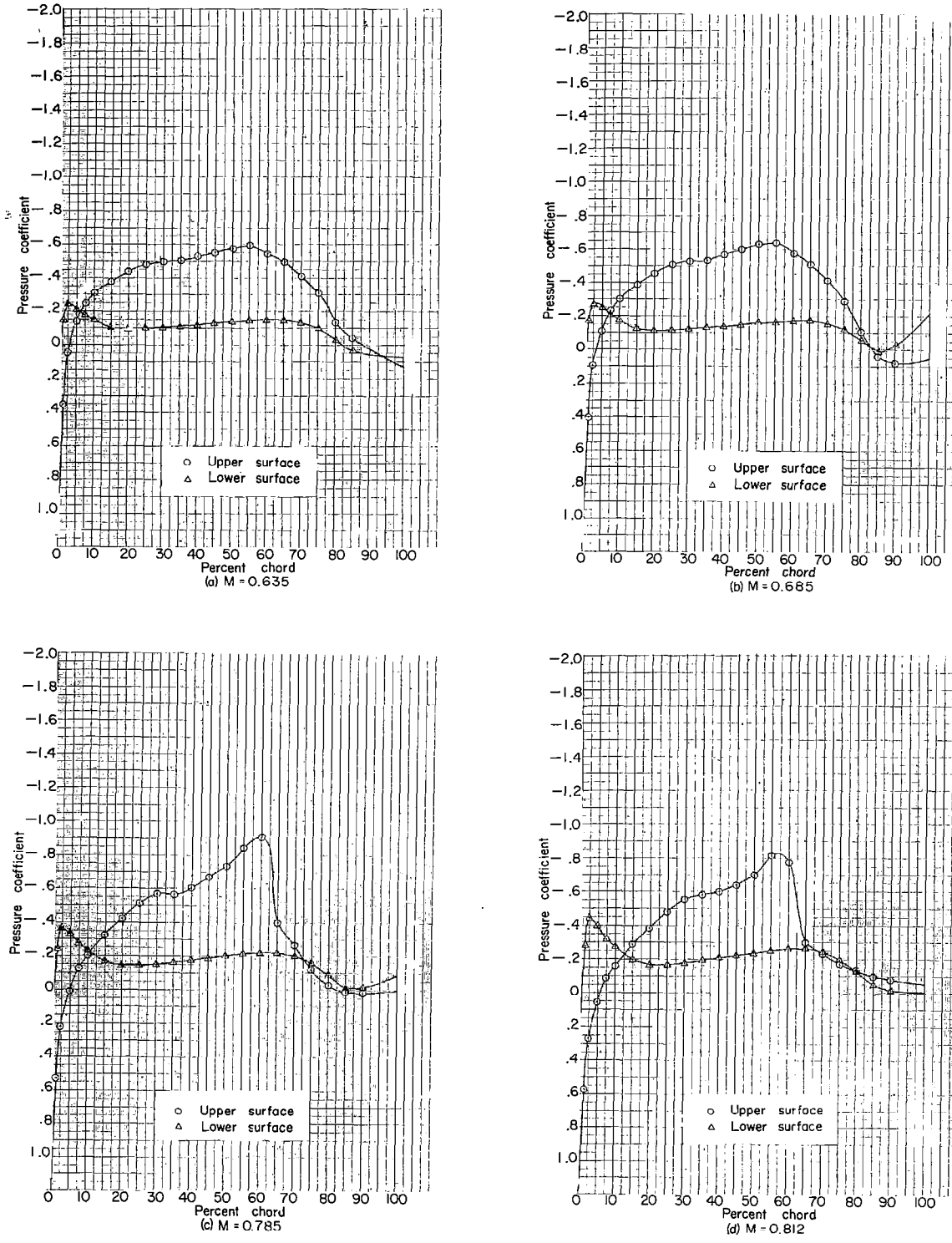
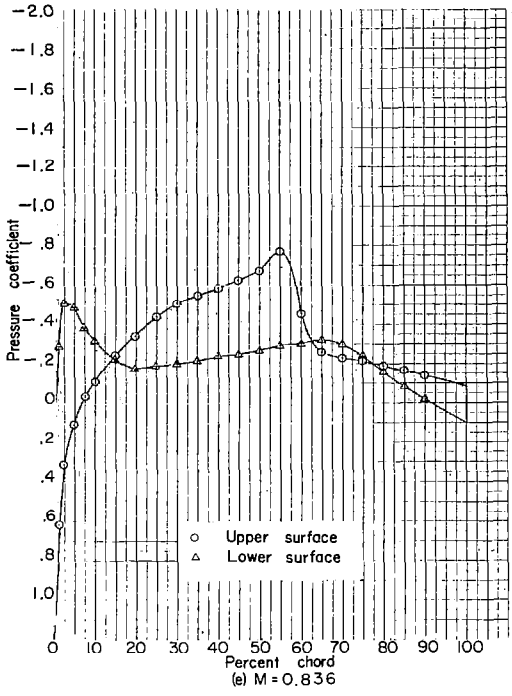
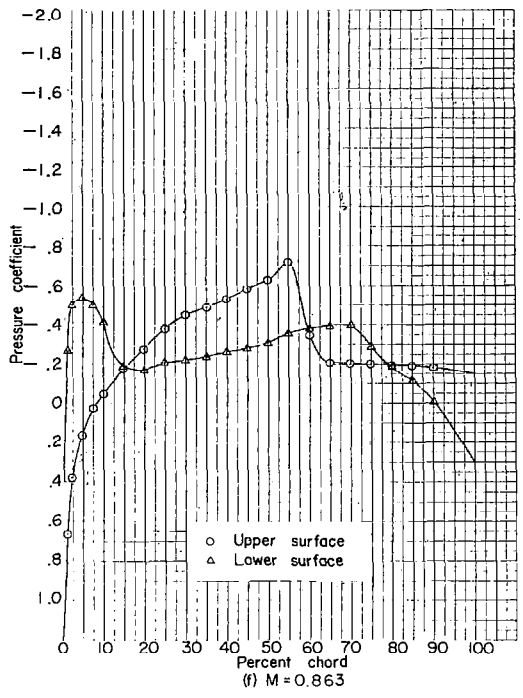
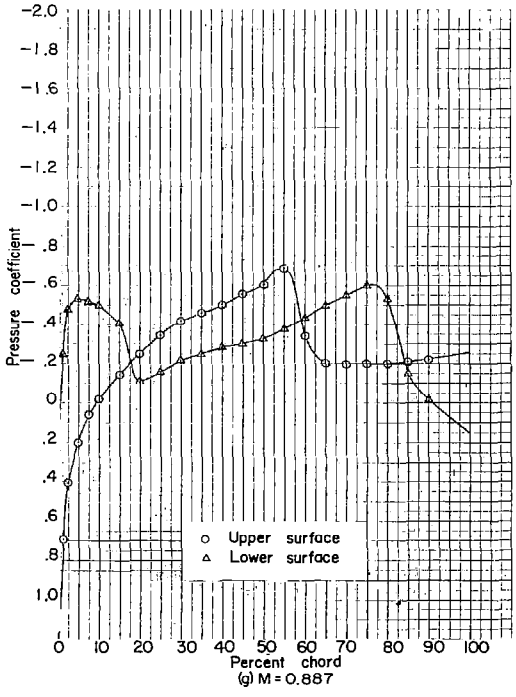


Figure 62.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 0^{\circ}$.







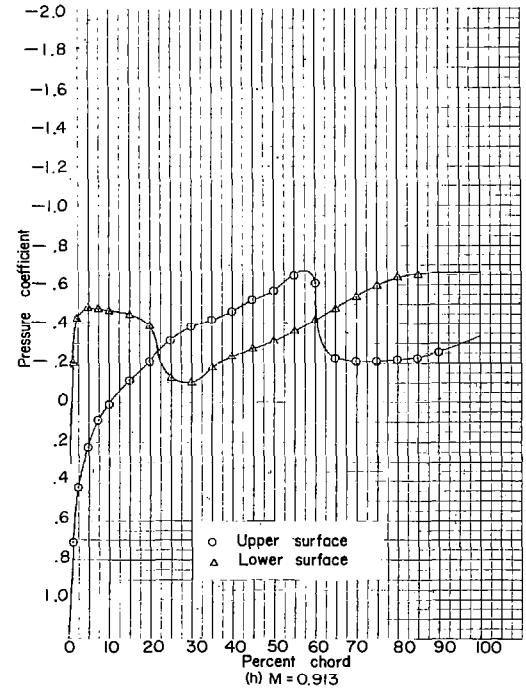
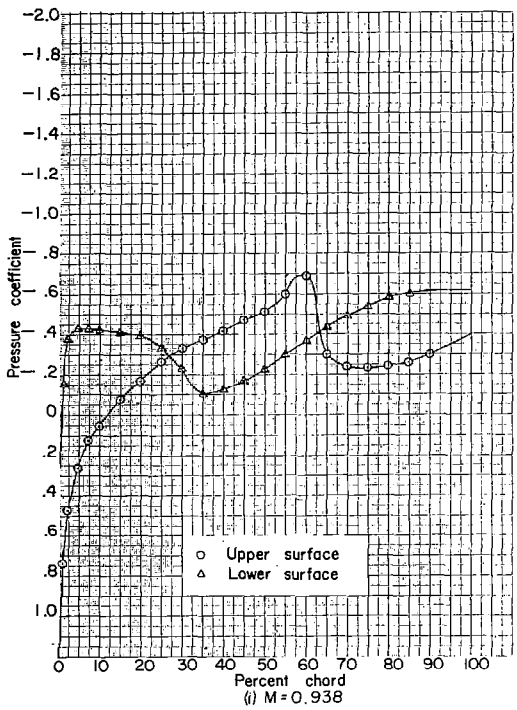
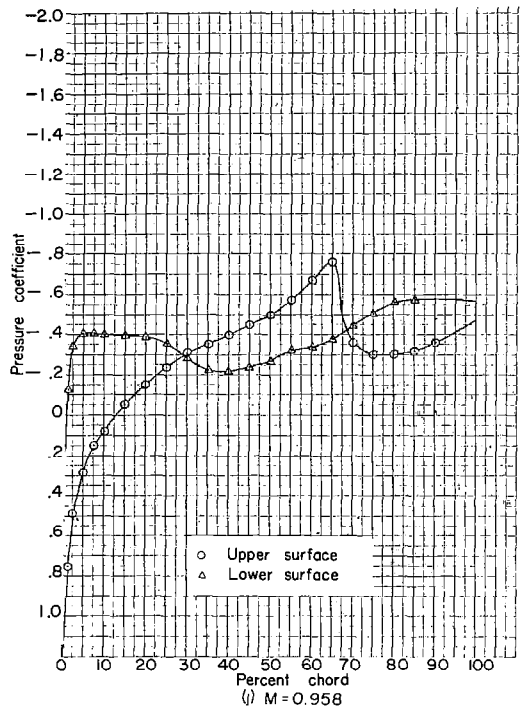


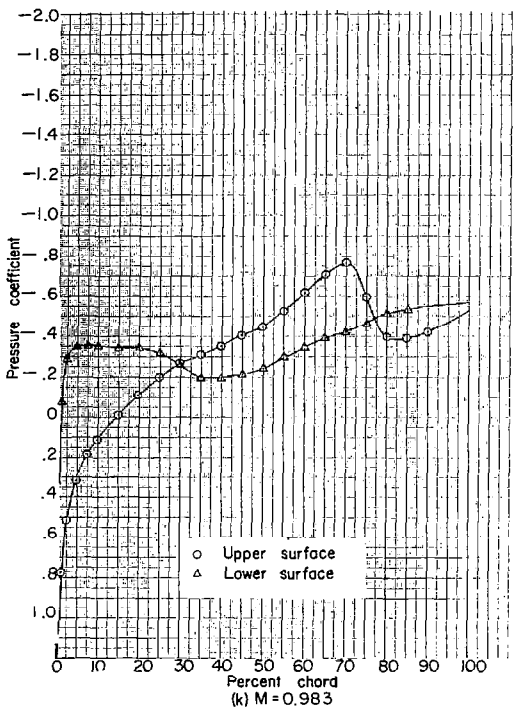
Figure 62.- Continued. NACA 16-512; $\alpha = 0^{\circ}$.



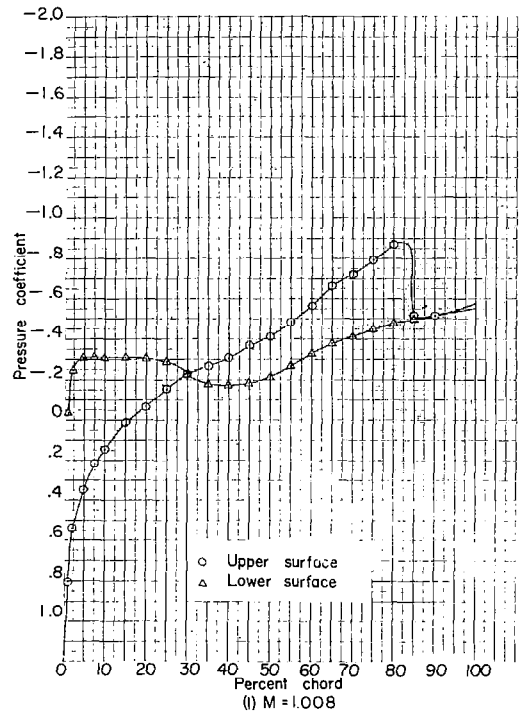
(i) M = 0.938



(j) M = 0.958

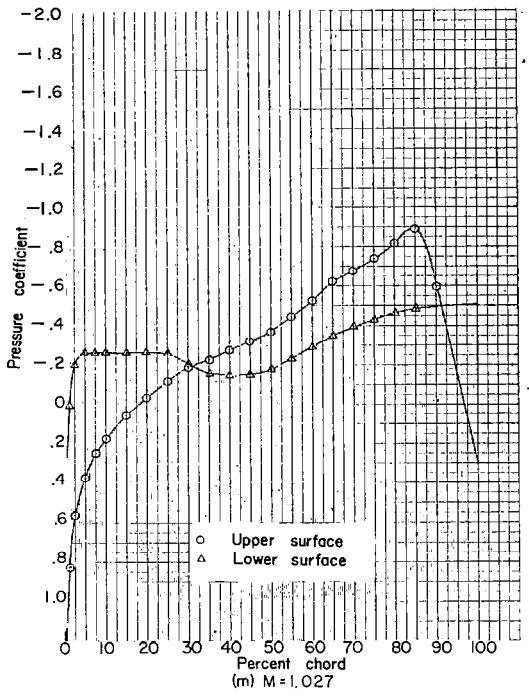


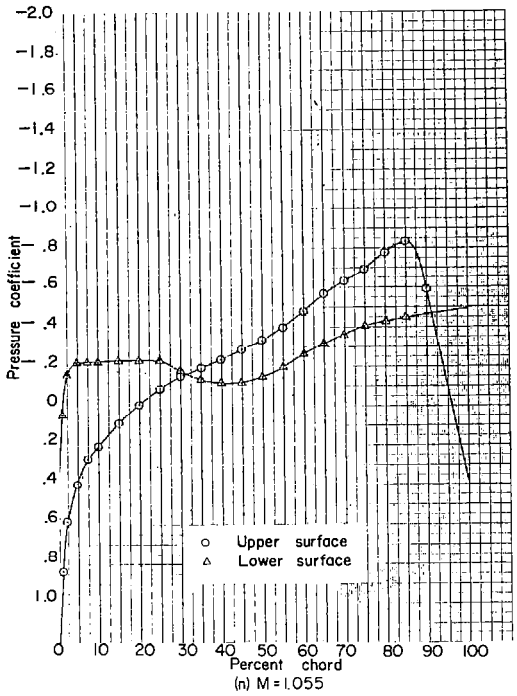
(k) M = 0.983

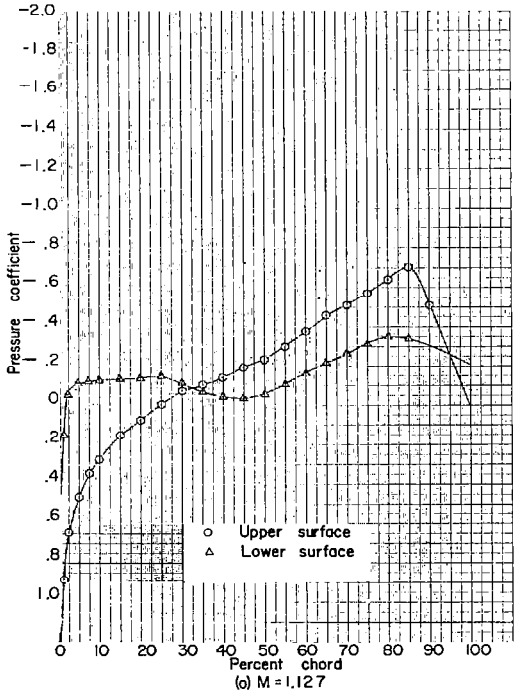


(l) M = 1.008

Figure 62.- Continued. NACA 16-512; $\alpha = 0^{\circ}$.







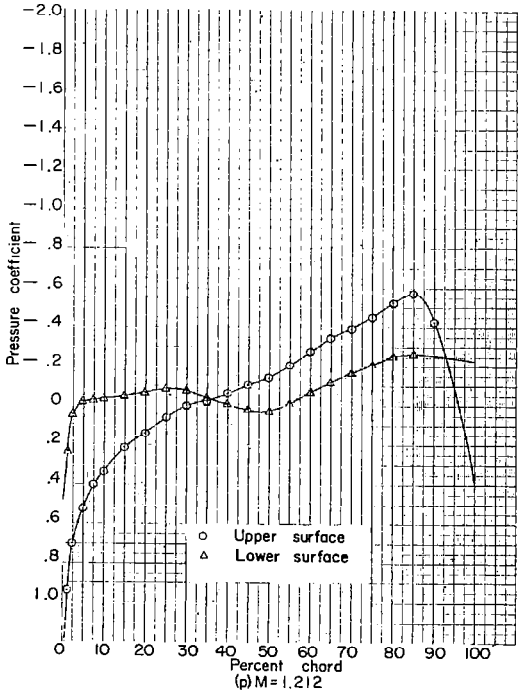


Figure 62.- Concluded. NACA 16-512; $\alpha = 0^{\circ}$.

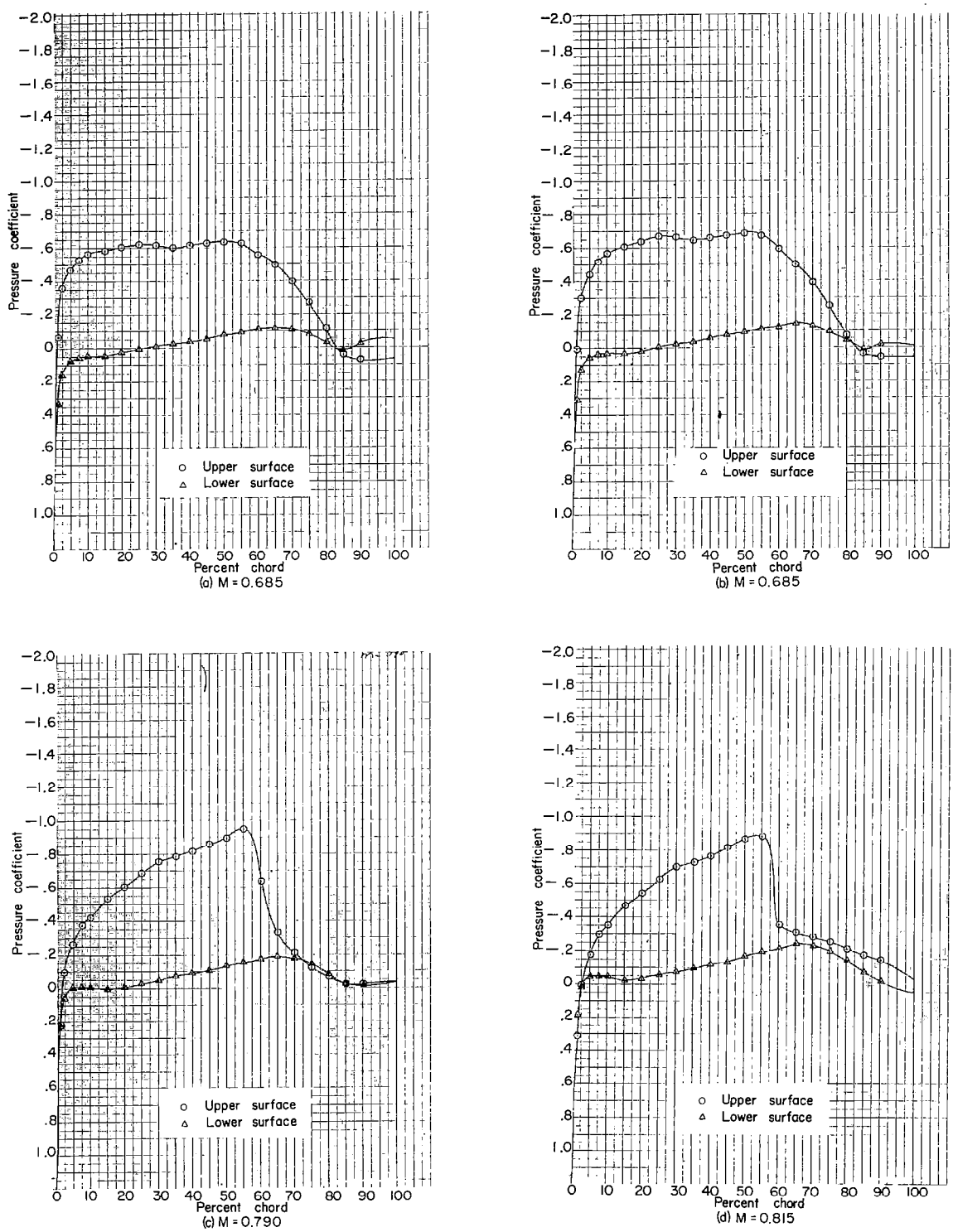
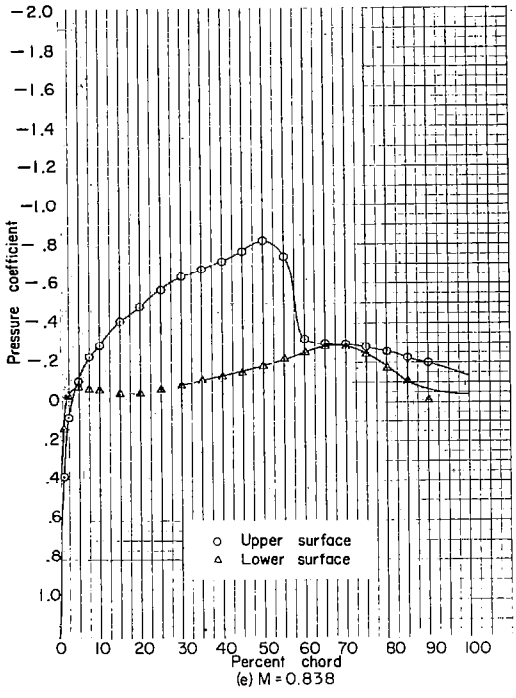
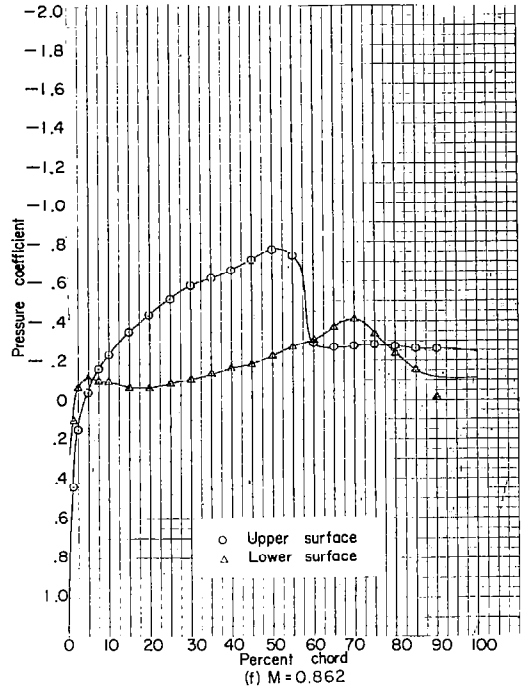
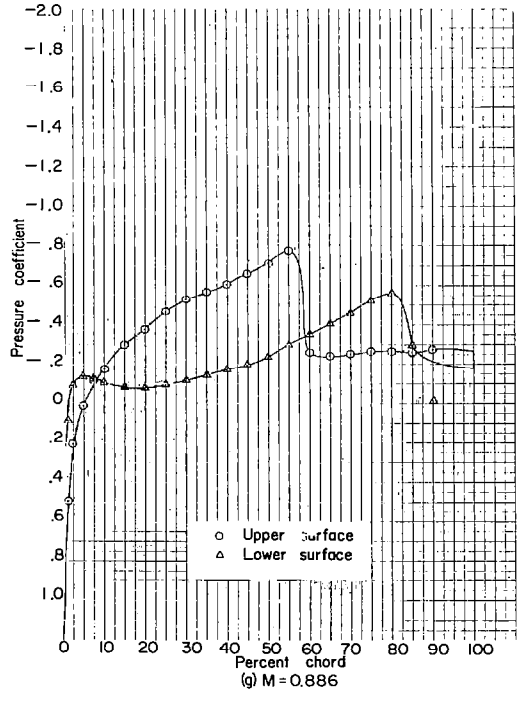


Figure 63.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 2^{\circ}$.







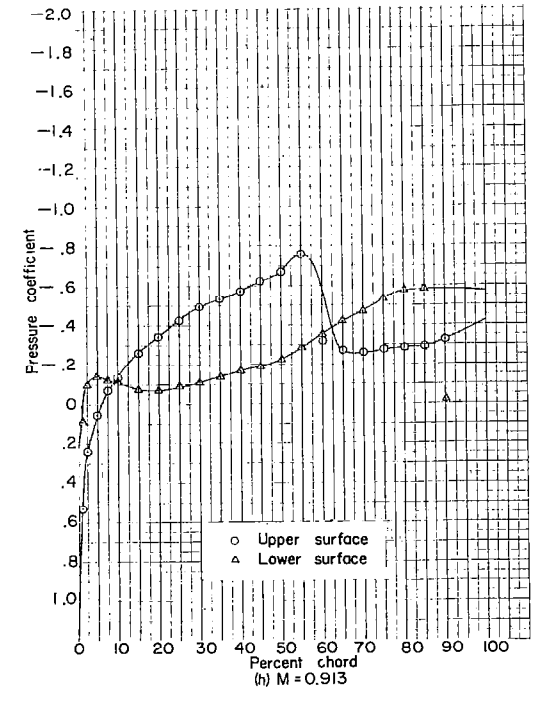
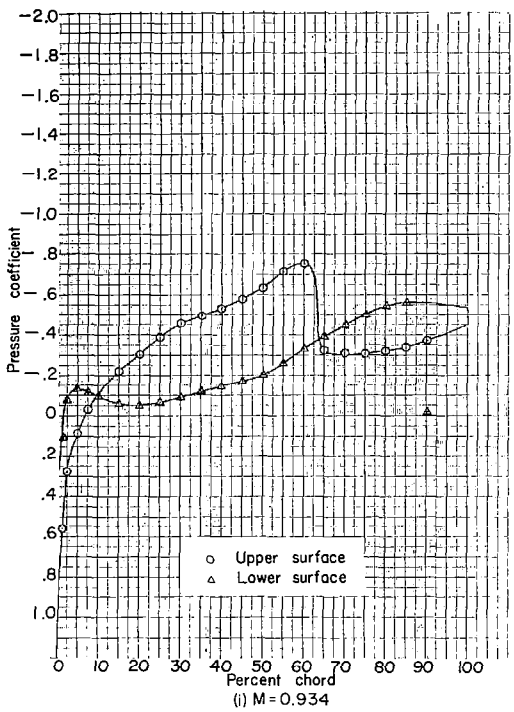
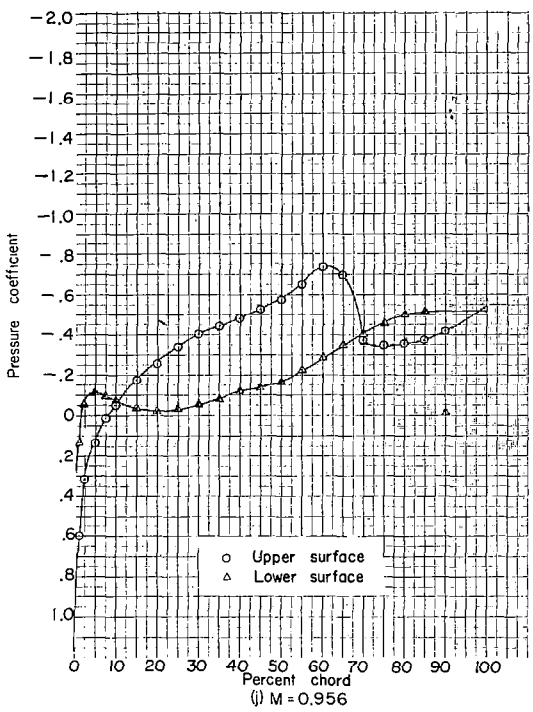
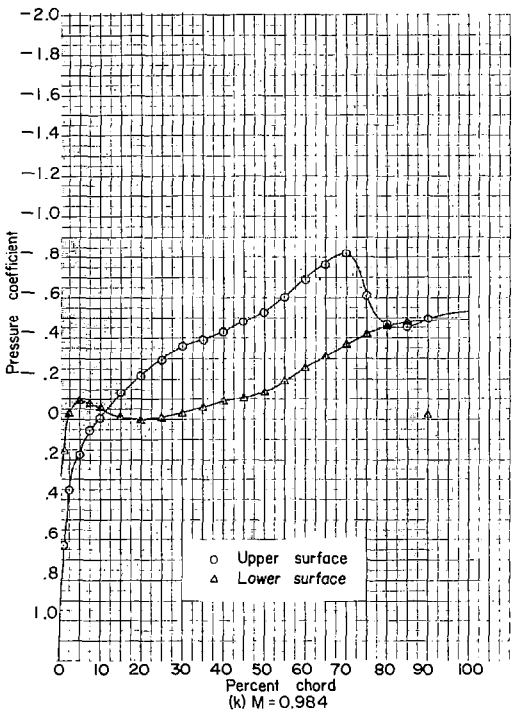


Figure 63.- Continued. NACA 16-512; $\alpha = 2^{\circ}$.







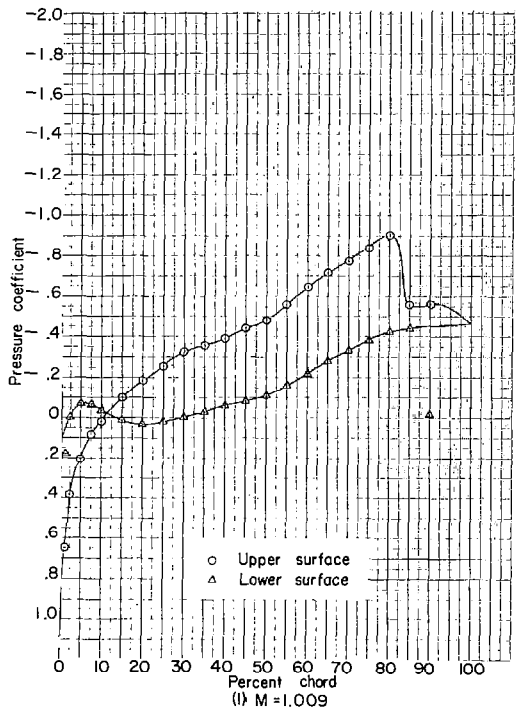
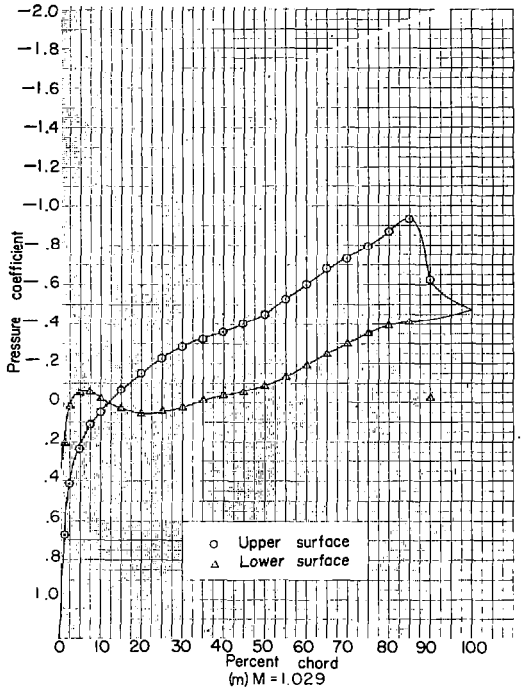
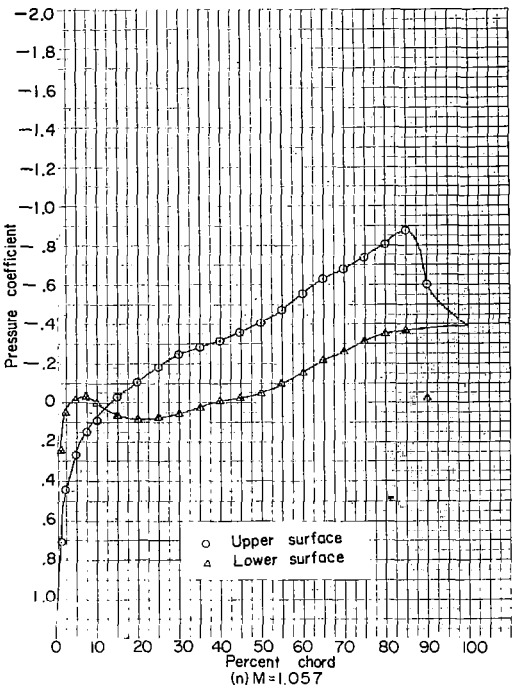
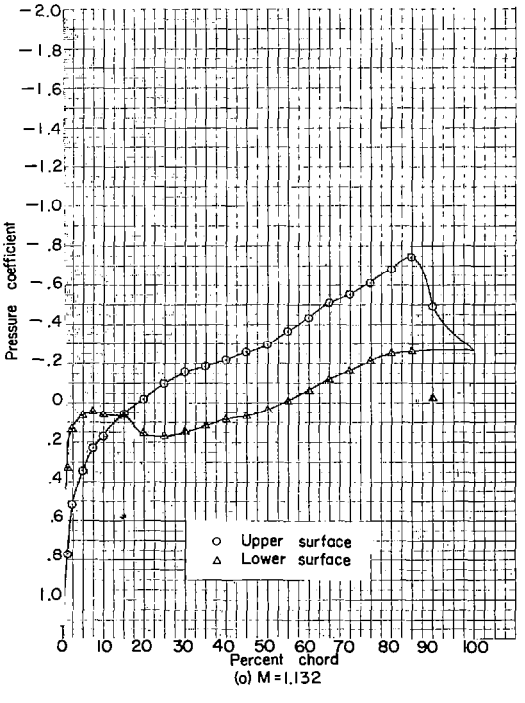


Figure 63.- Continued. NACA 16-512; $\alpha = 2^{\circ}$.







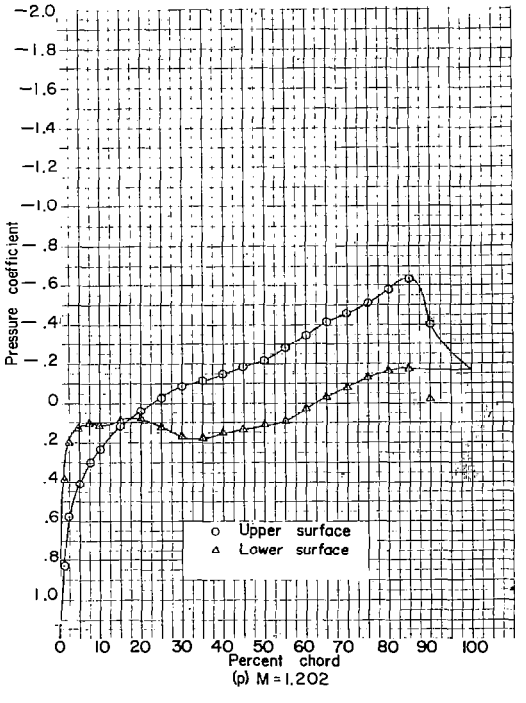


Figure 63.- Concluded. NACA 16-512; $\alpha = 2^{\circ}$.

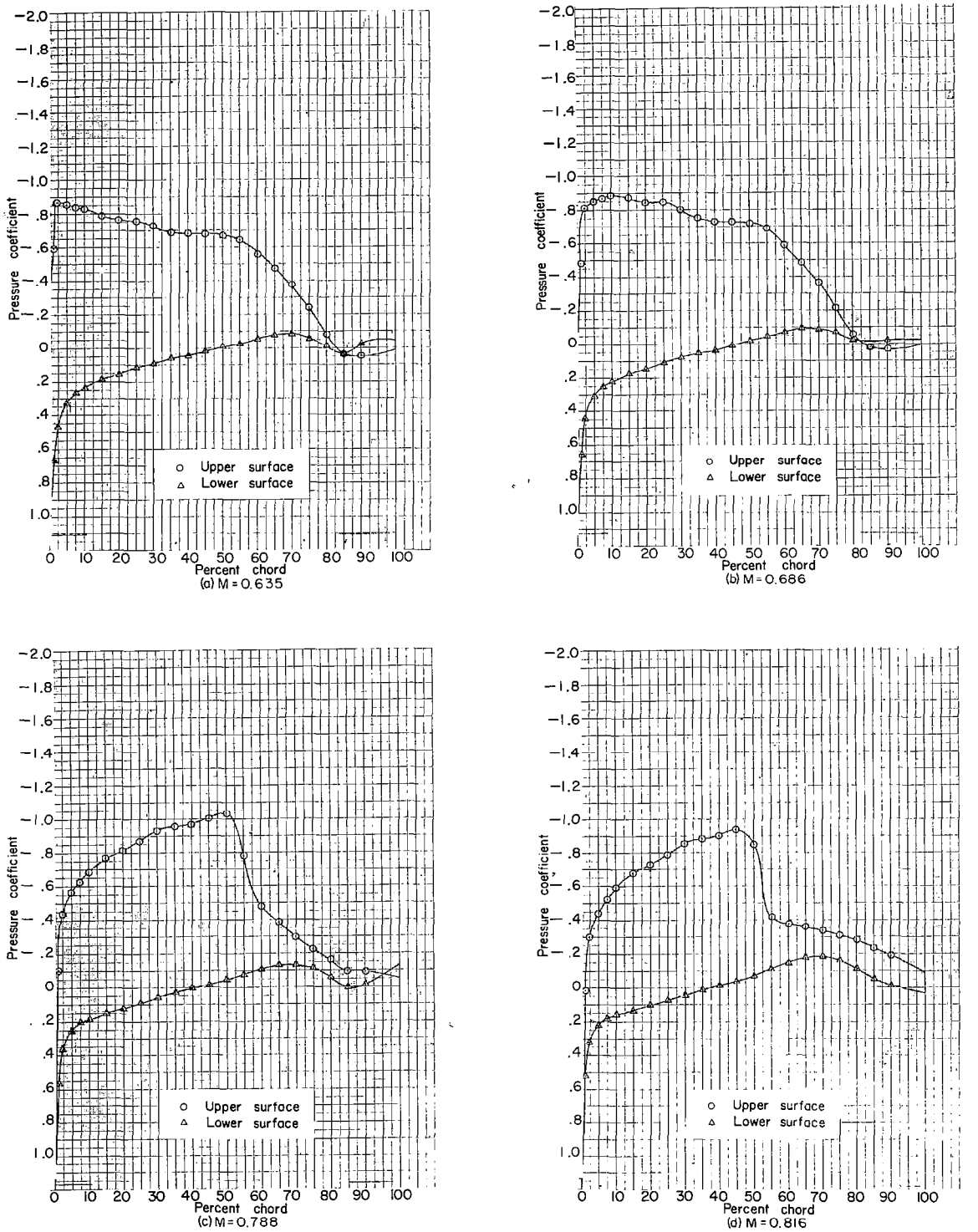
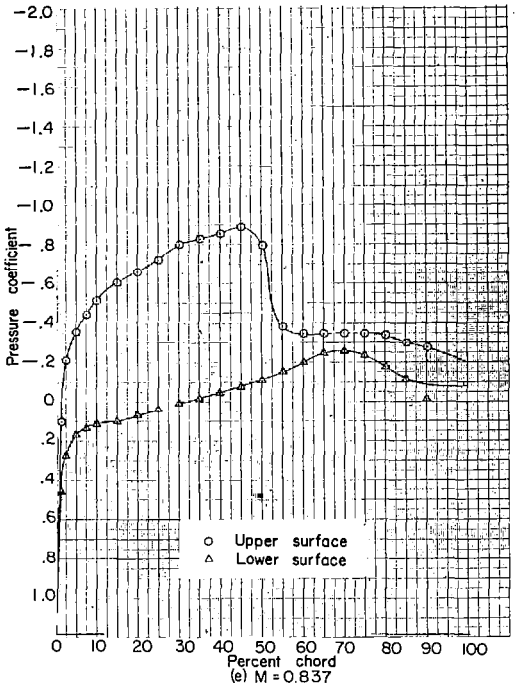
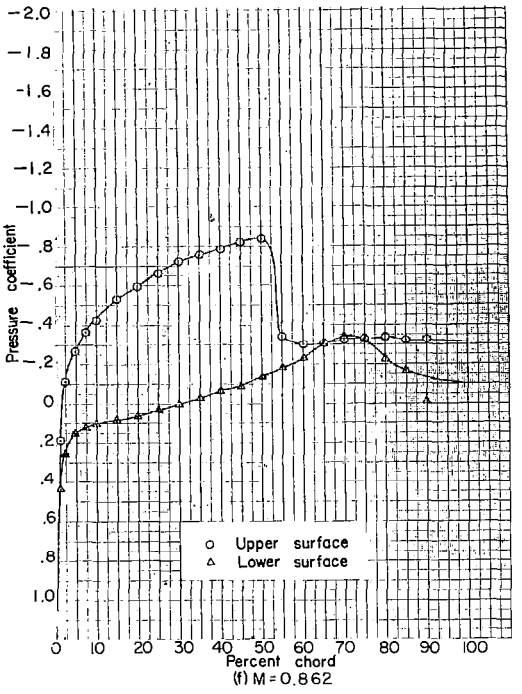
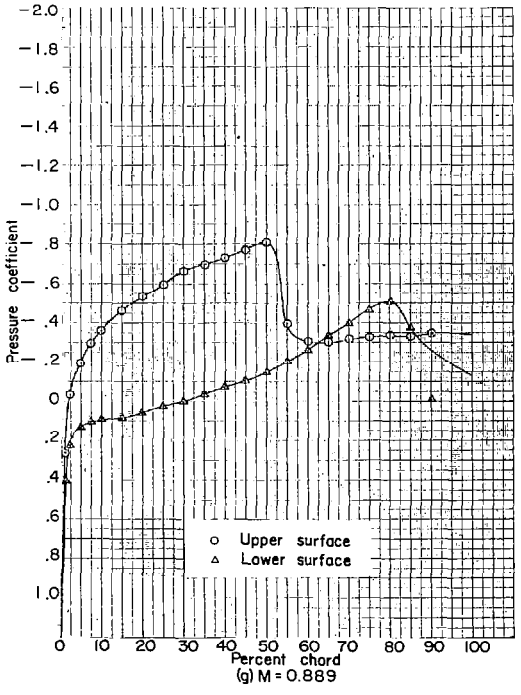


Figure 64.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 4^{\circ}$.







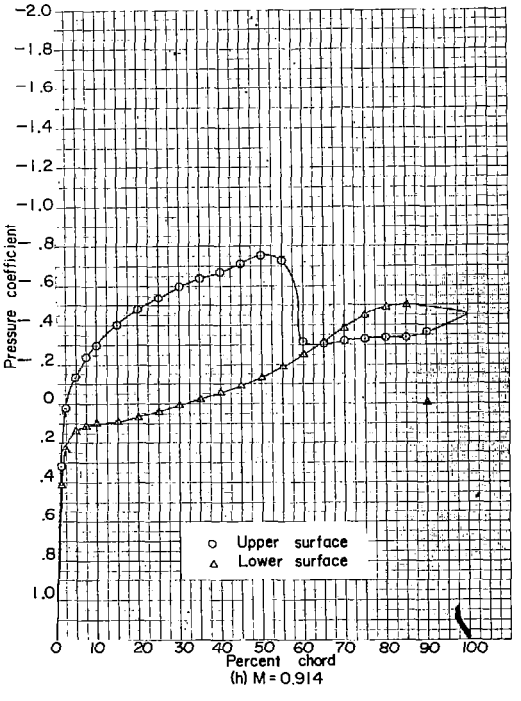
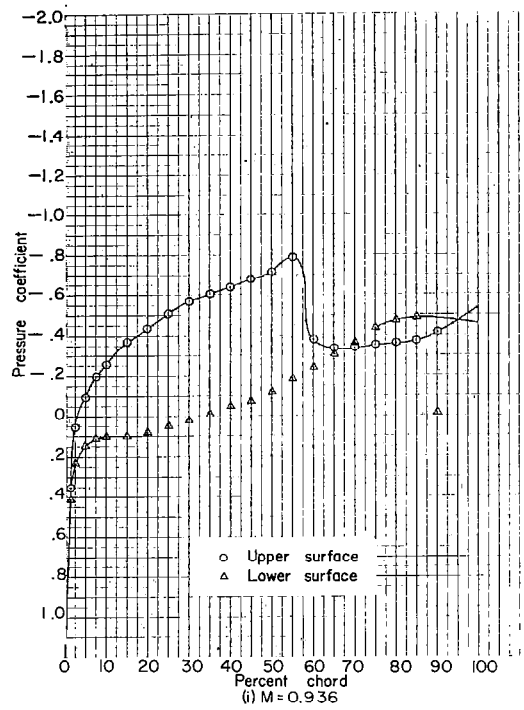
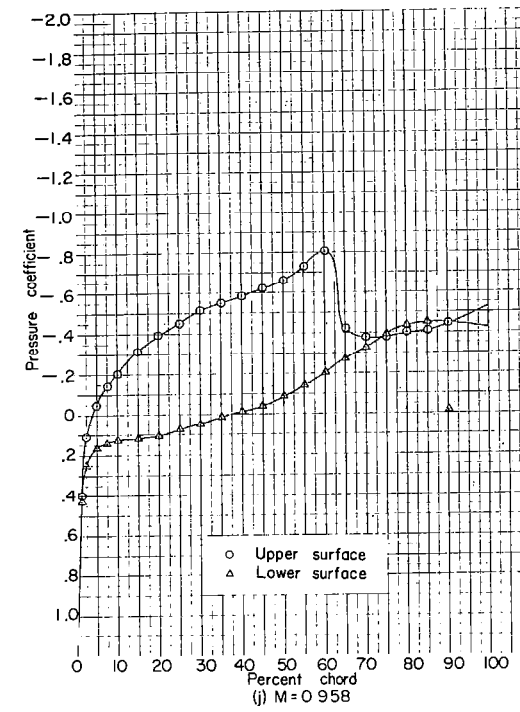
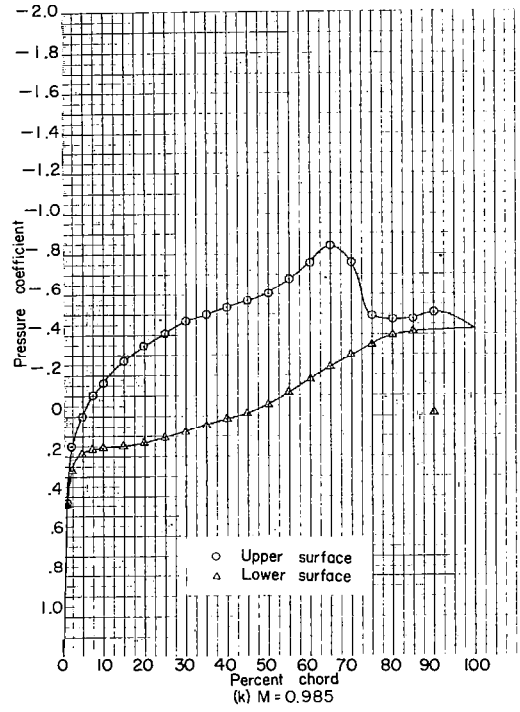


Figure 64.- Continued. NACA 16-512; $\alpha = 4^{\circ}$.







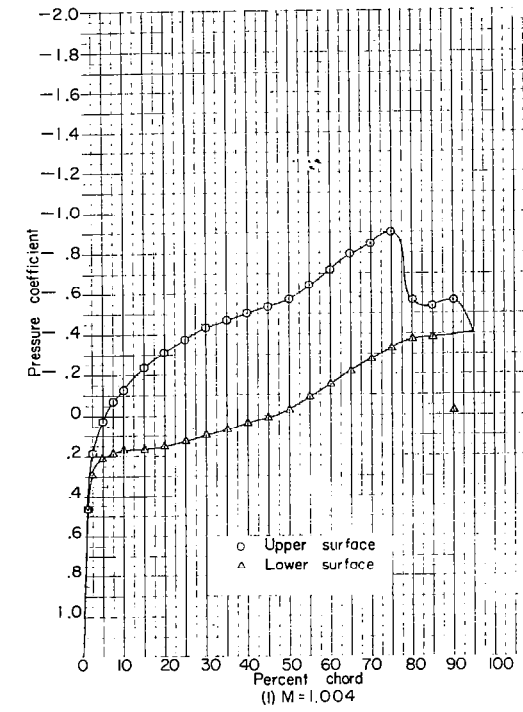
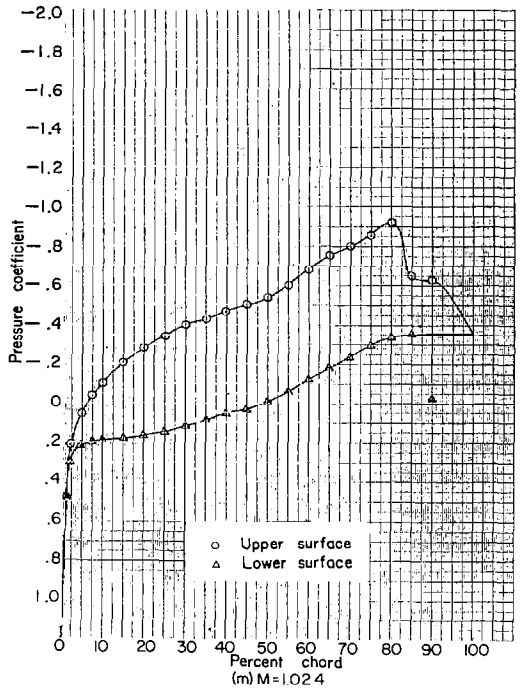
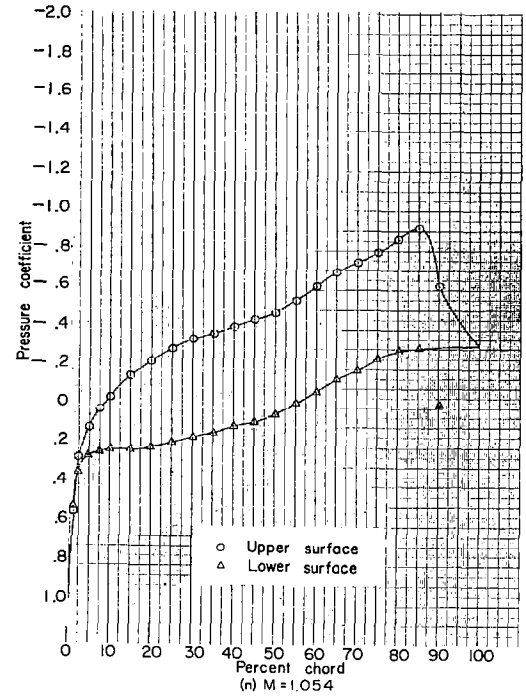
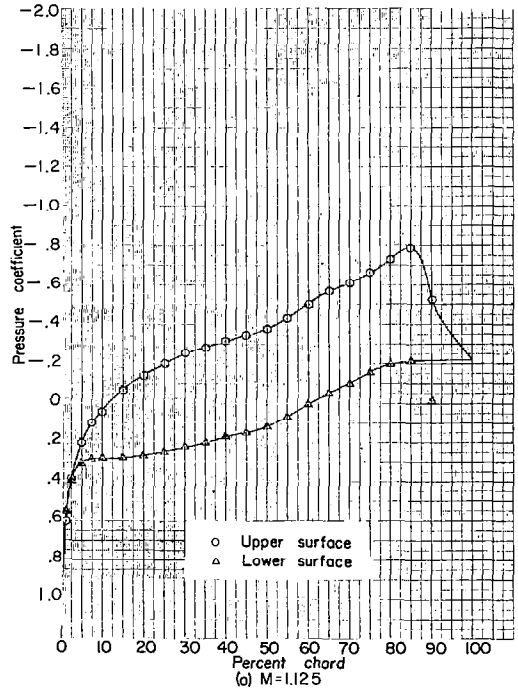


Figure 64.- Continued. NACA 16-512; $\alpha = 4^{\circ}$.







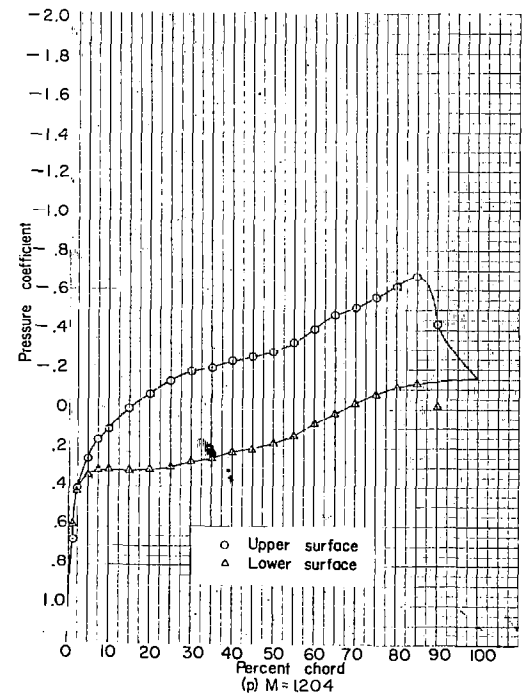


Figure 64.- Concluded. NACA 16-512; $\alpha = 4^{\circ}$.

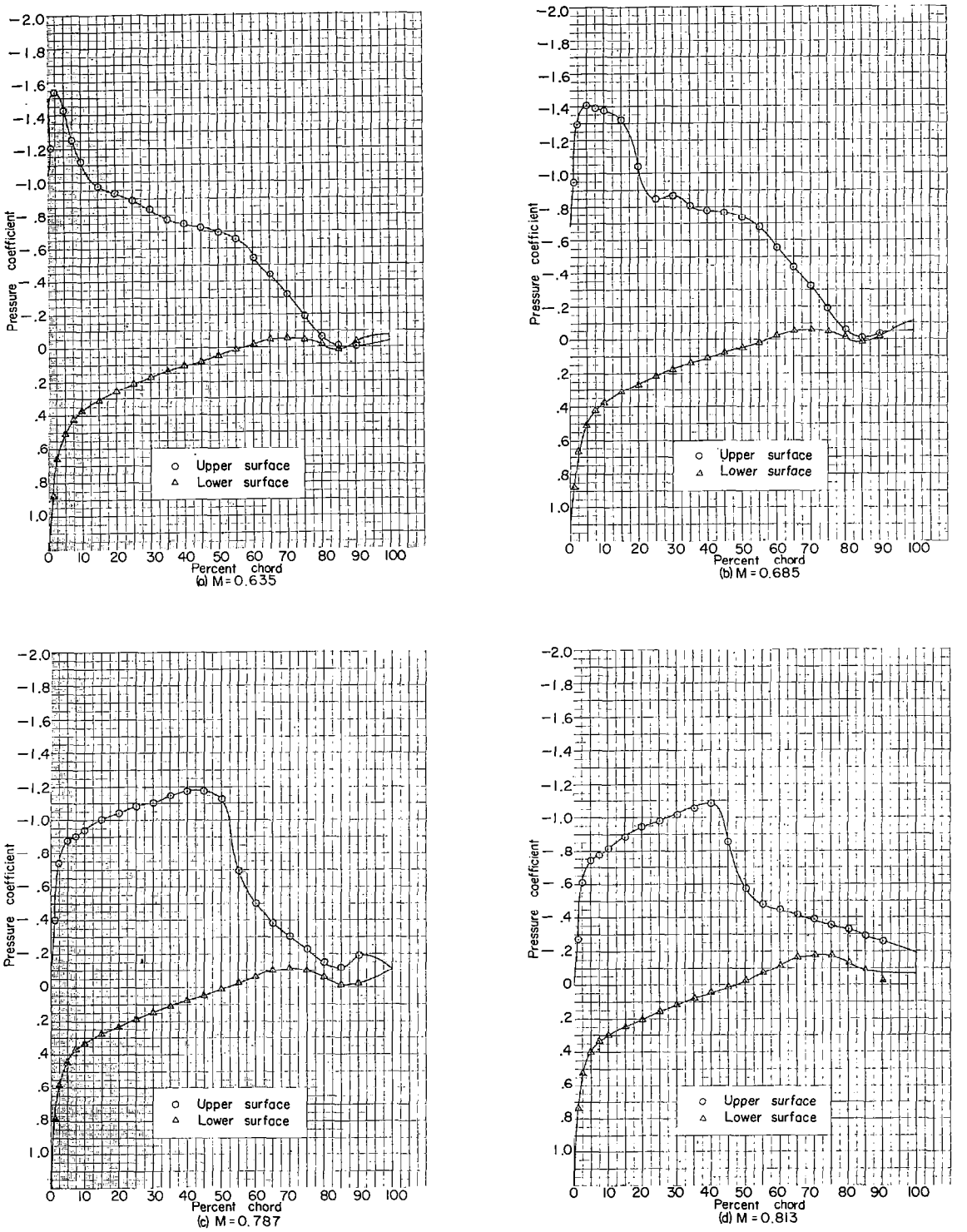
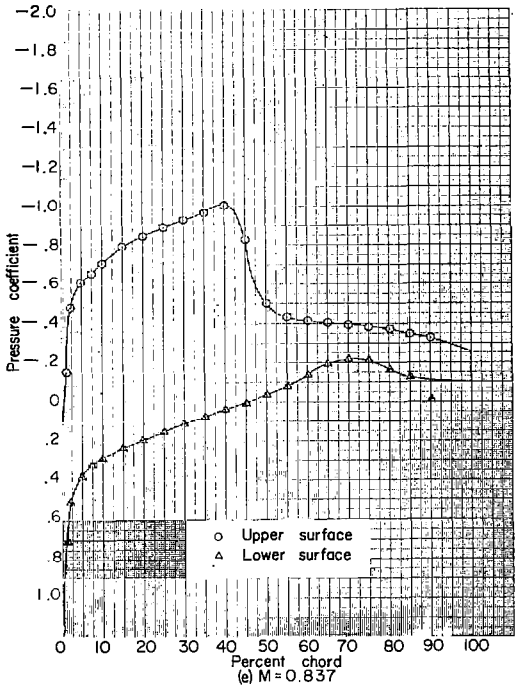
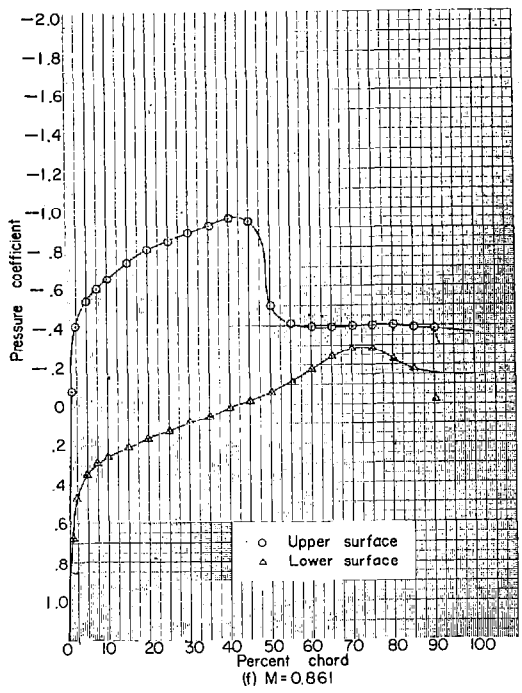
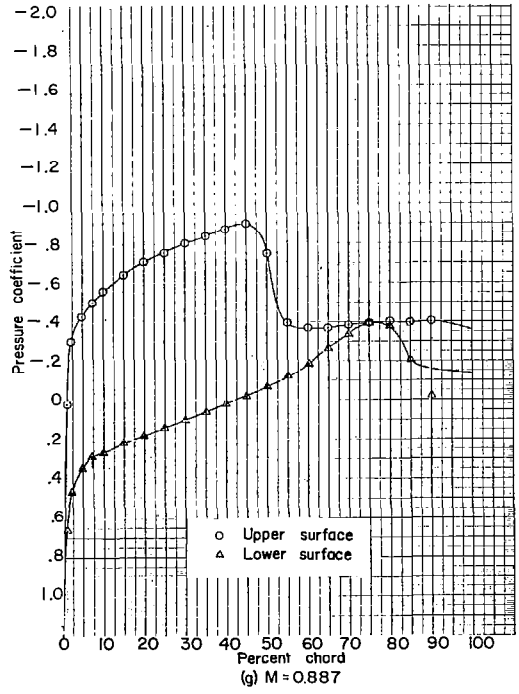


Figure 65.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 6^{\circ}$.







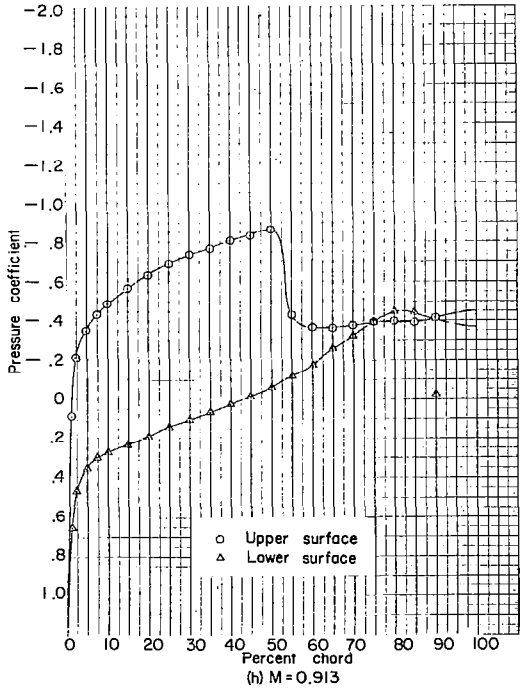
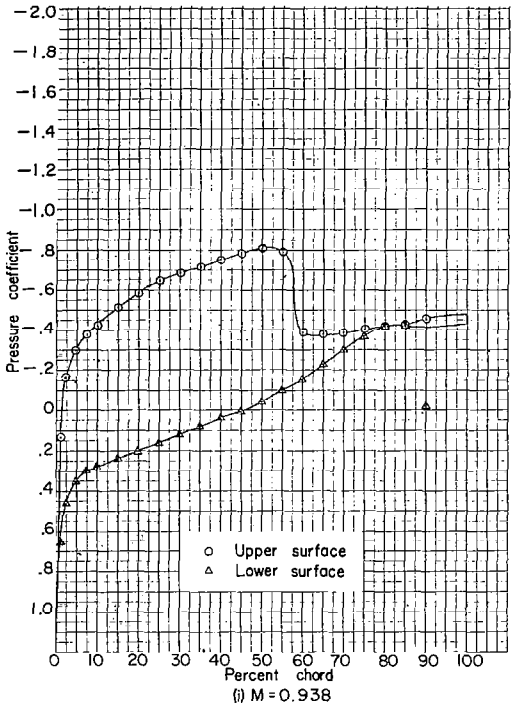
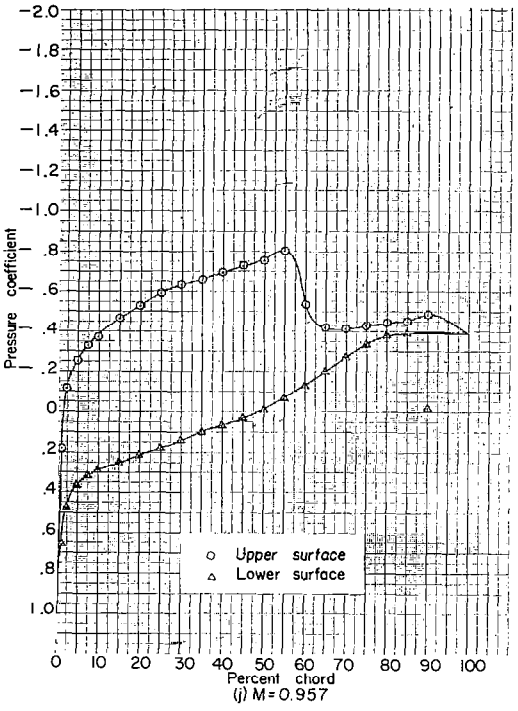
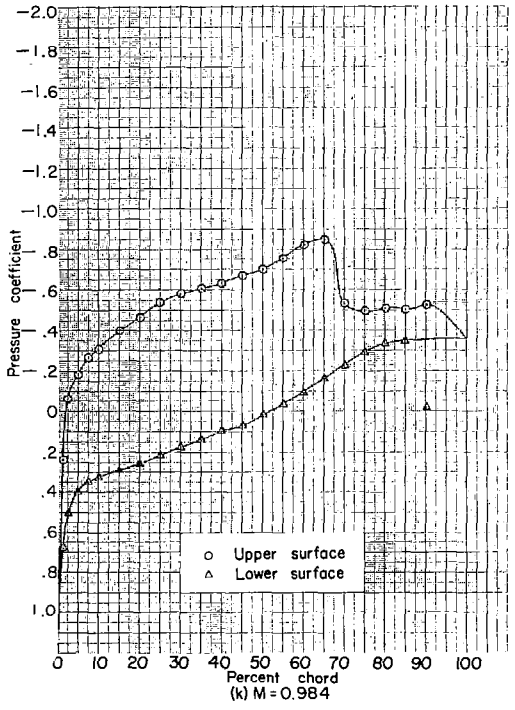


Figure 65.- Continued. NACA 16-512; $\alpha = 6^{\circ}$.







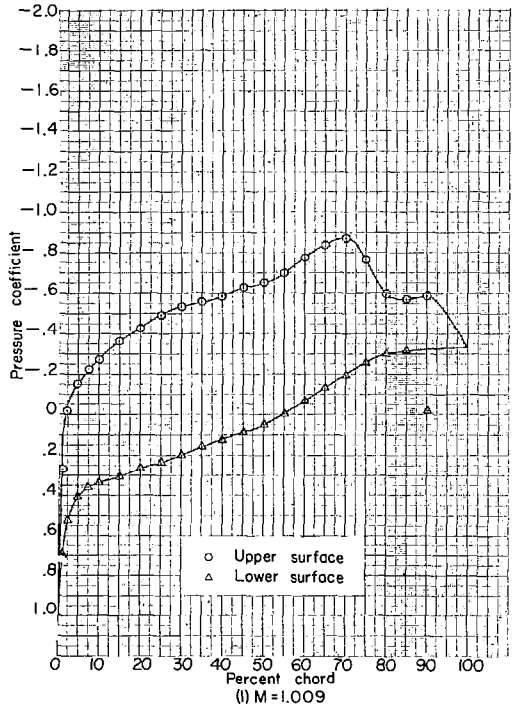
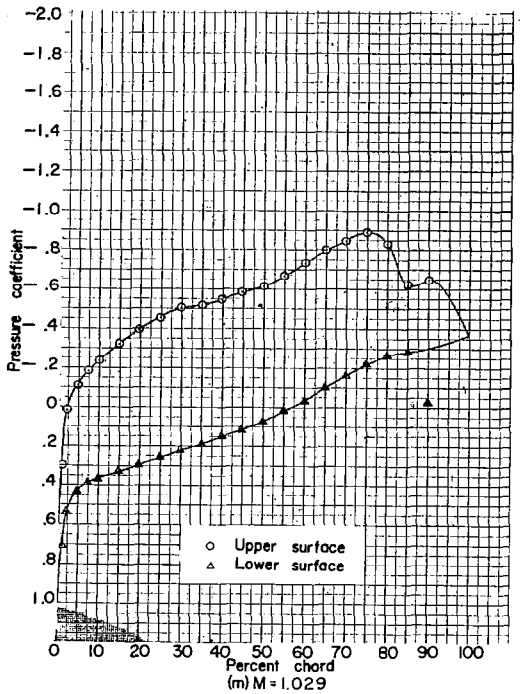
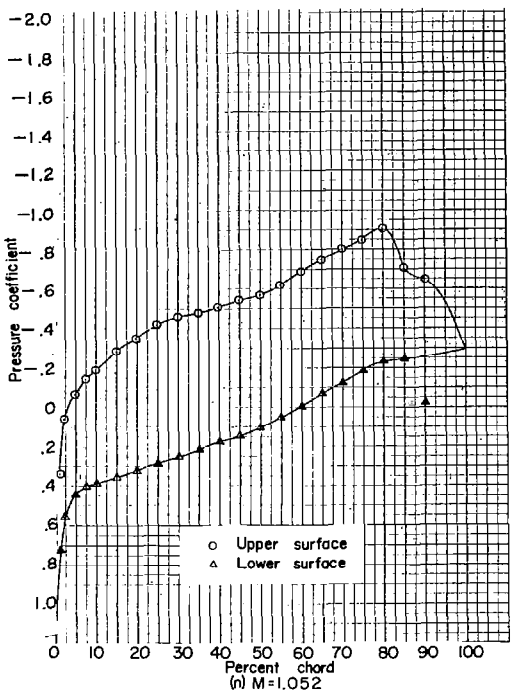
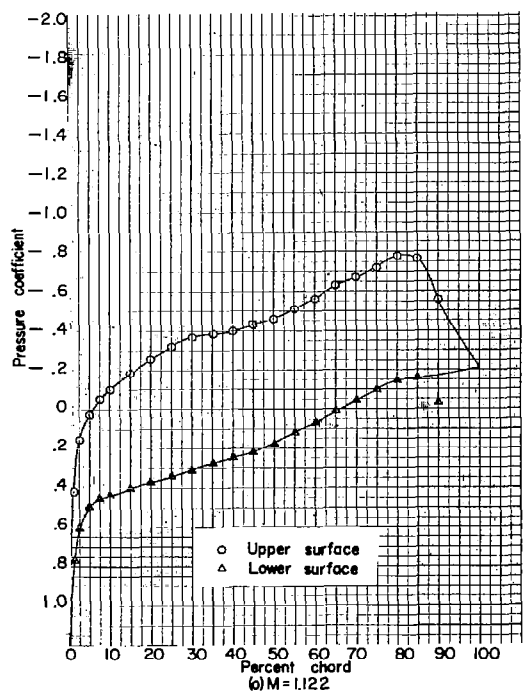


Figure 65.- Continued. NACA 16-512; $\alpha = 6^{\circ}$.







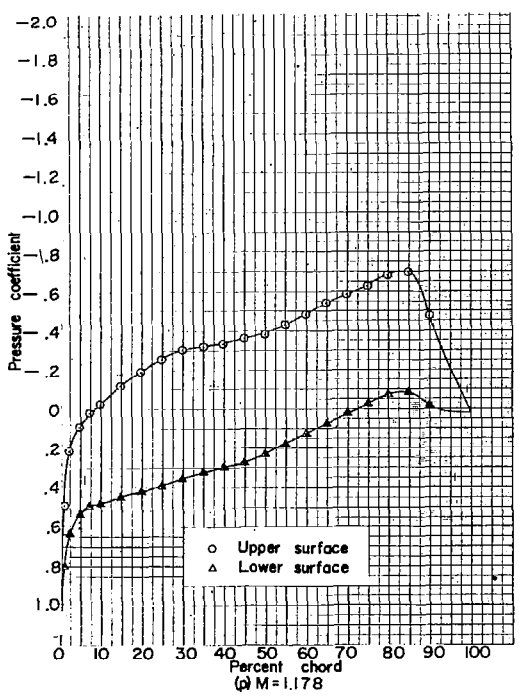


Figure 65.- Concluded. NACA 16-512; $\alpha = 6^{\circ}$.

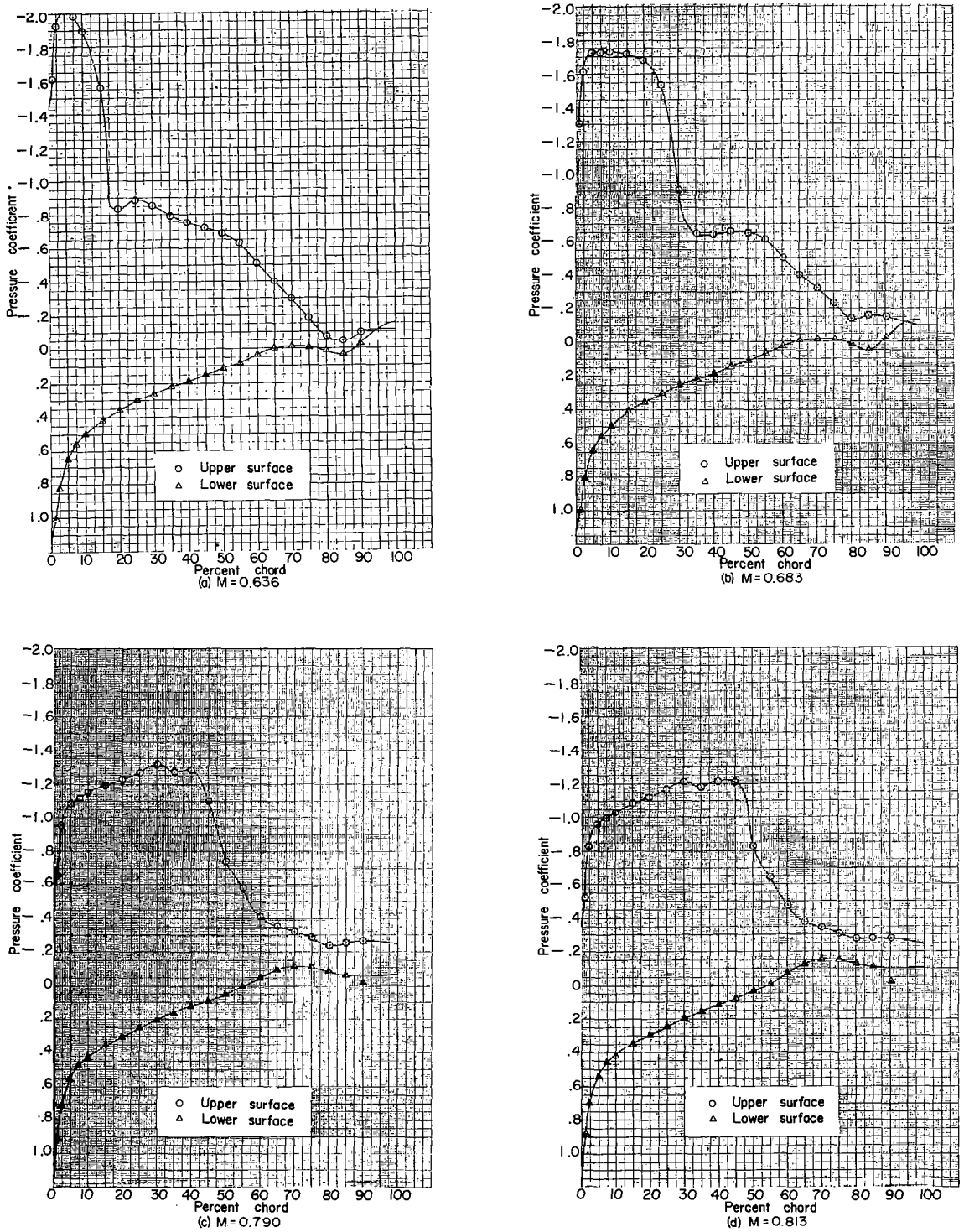
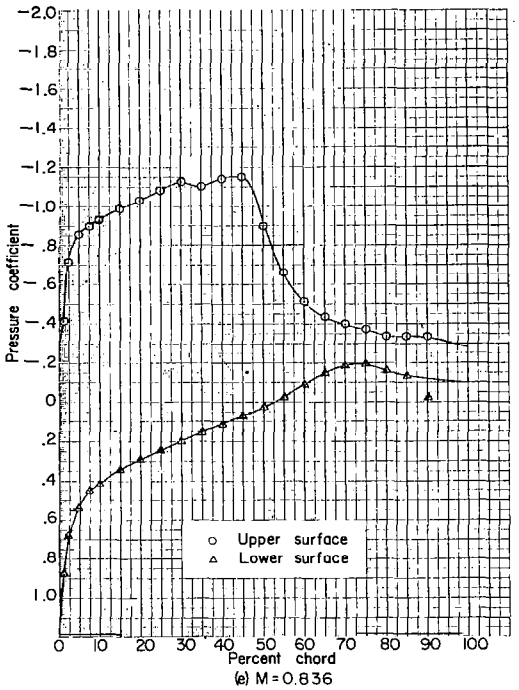
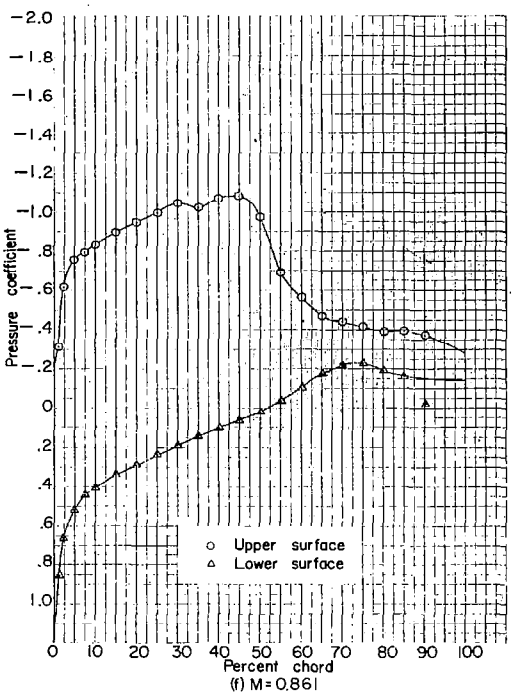
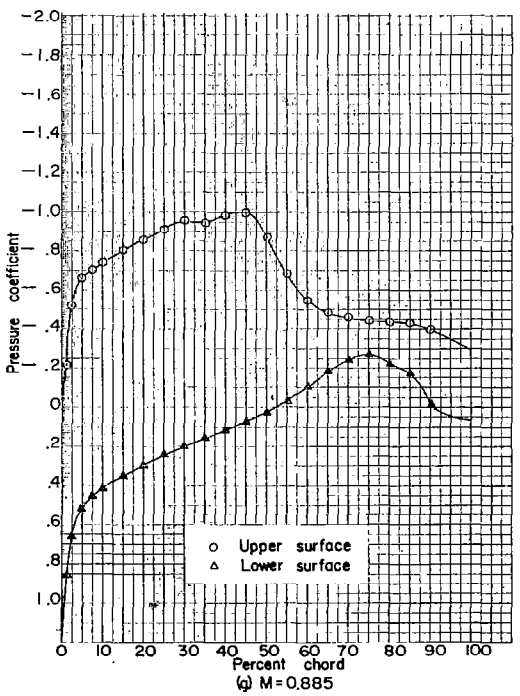


Figure 66.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 8^{\circ}$.







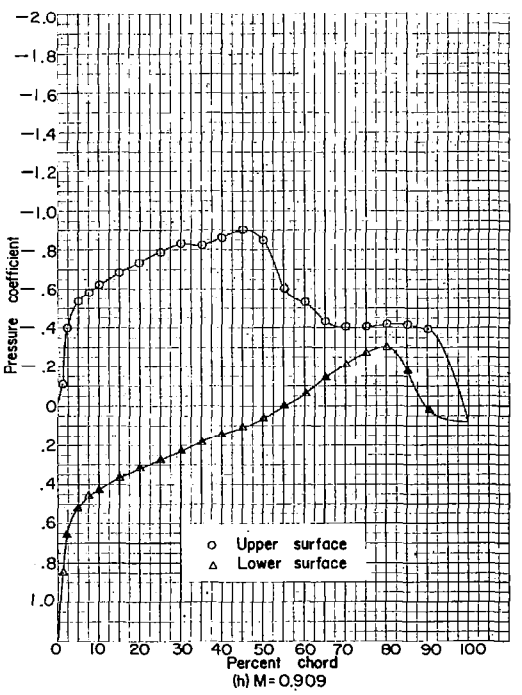
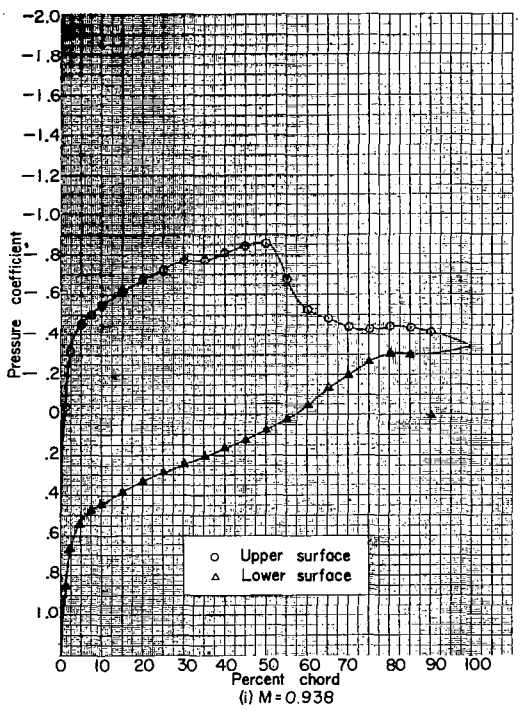
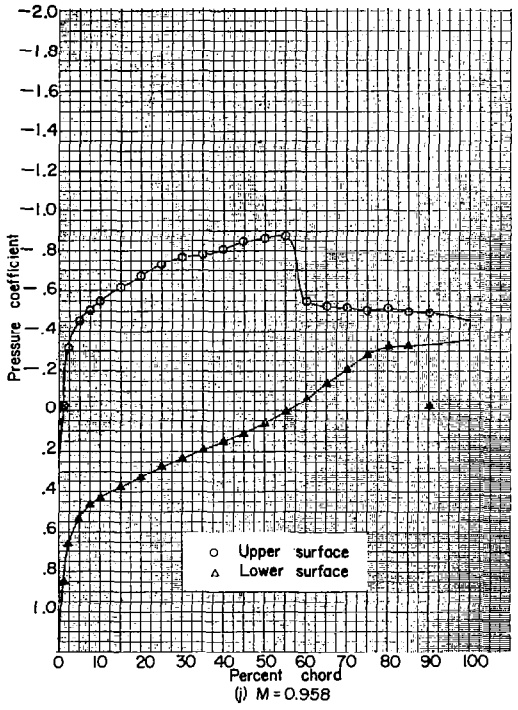
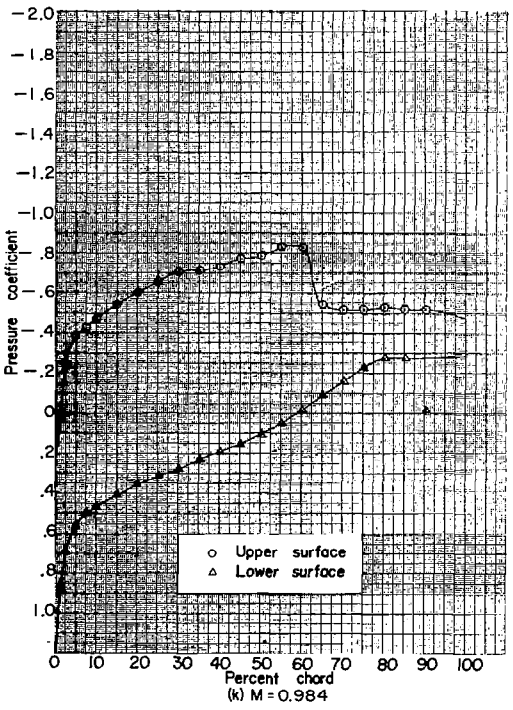


Figure 66.- Continued. NACA 16-512; $\alpha = 8^{\circ}$.







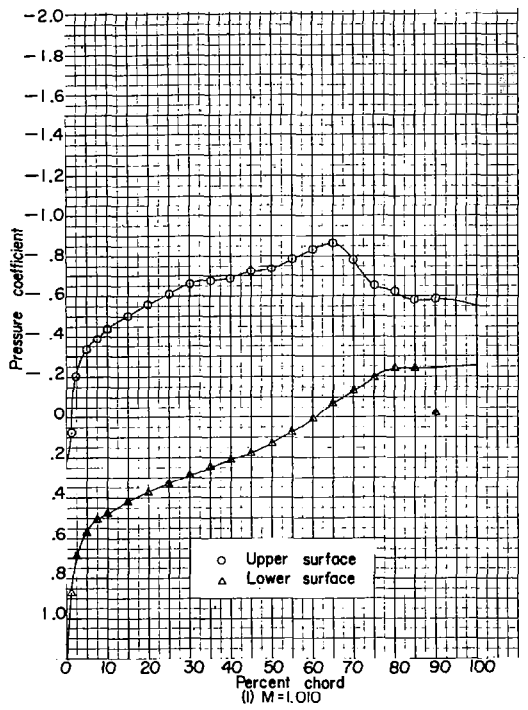
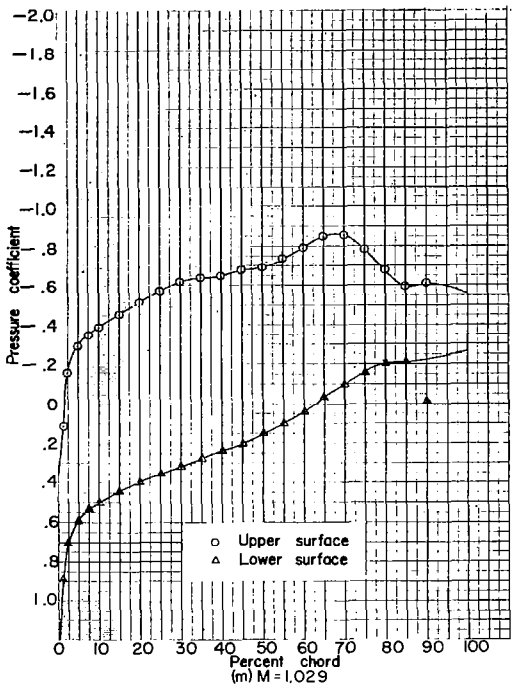
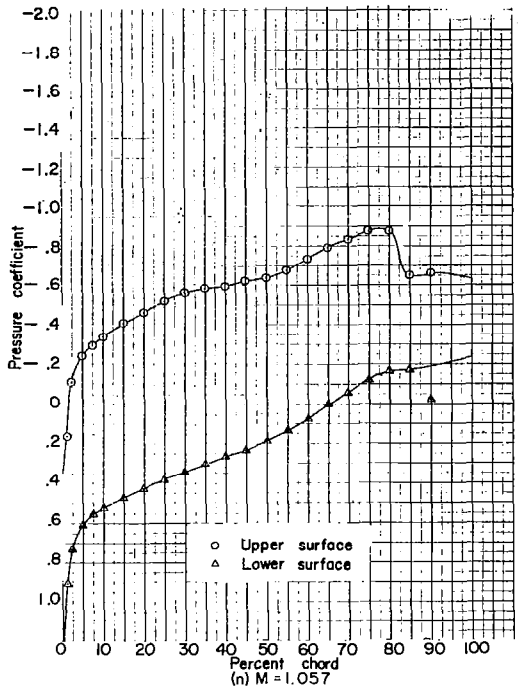
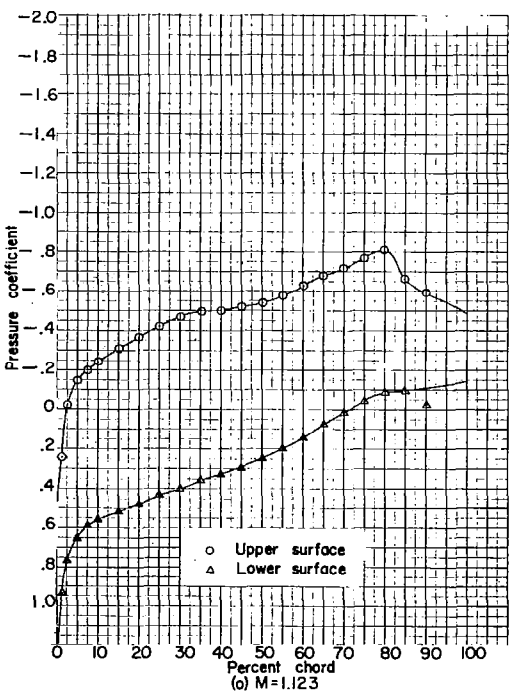


Figure 66.- Continued. NACA 16-512; $\alpha = 8°$.







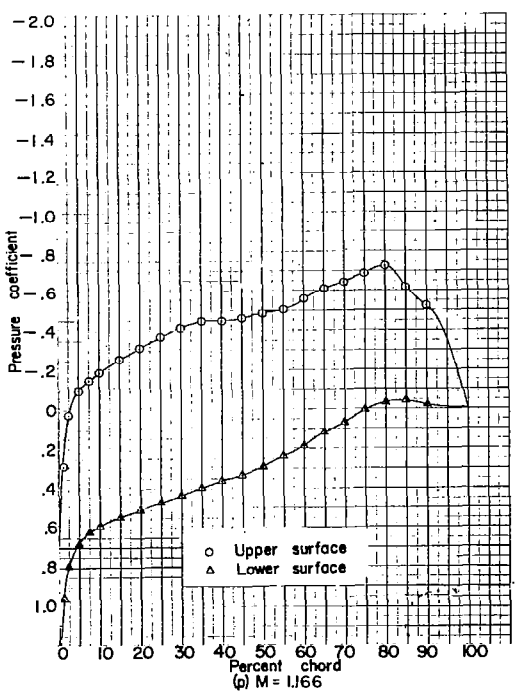


Figure 66.- Concluded. NACA 16-512; $\alpha = 8^{\circ}$.

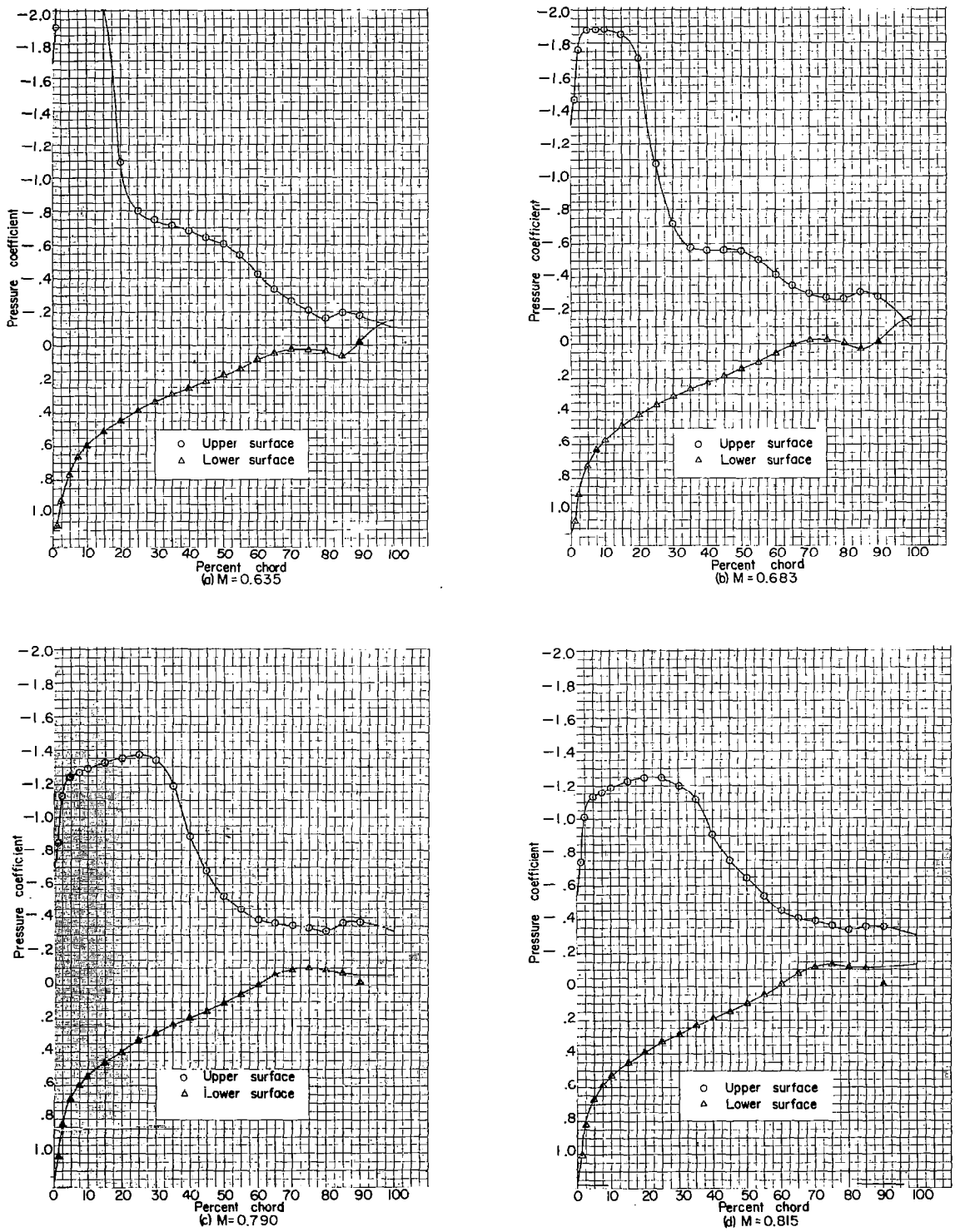
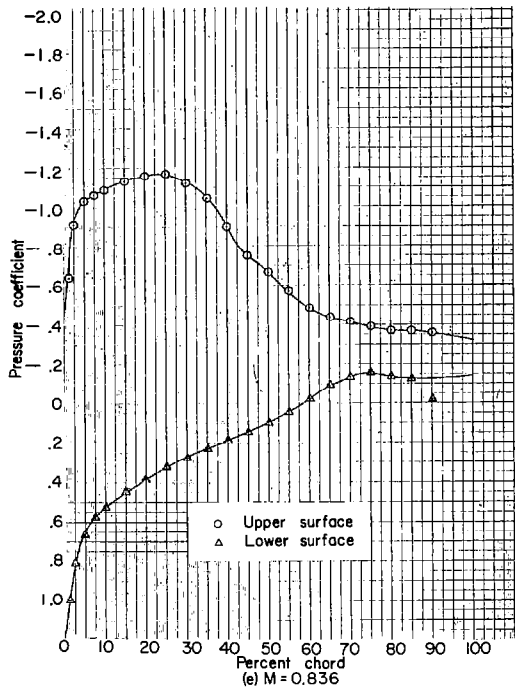
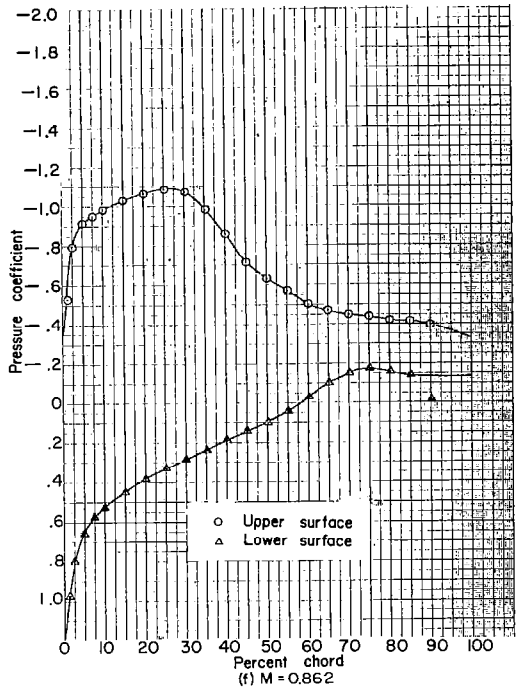
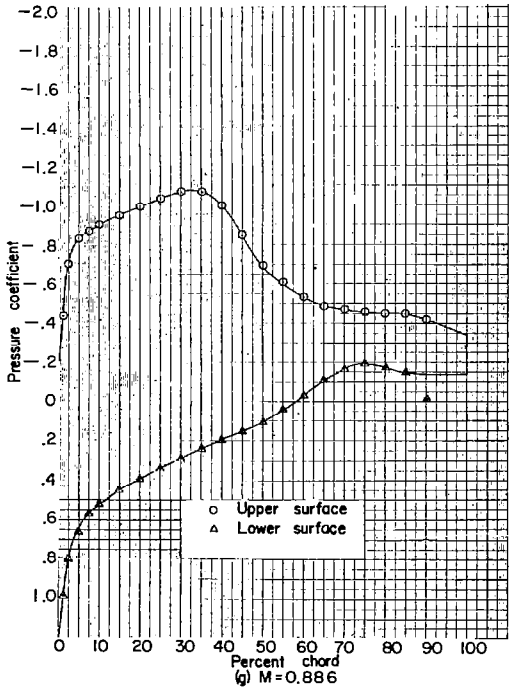


Figure 67.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 10^{\circ}$.







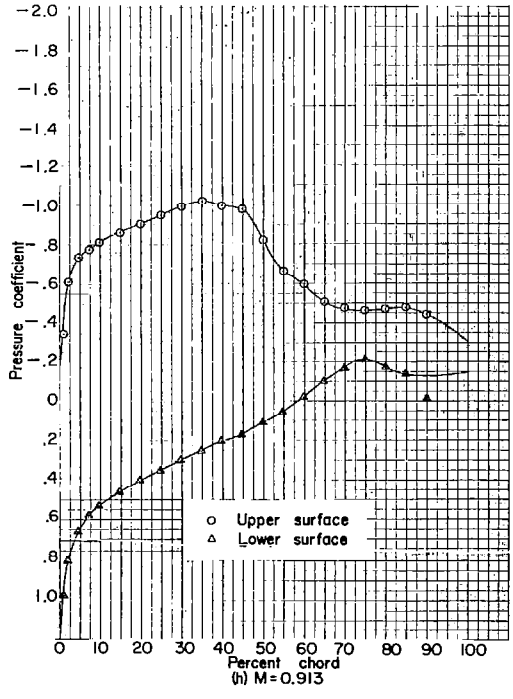
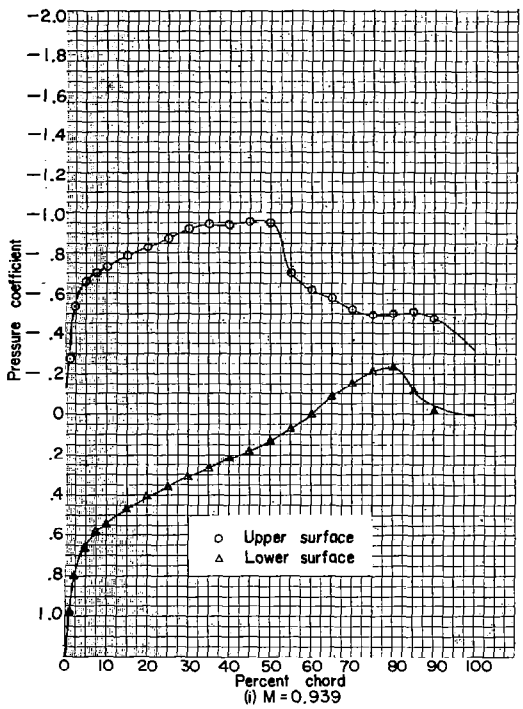
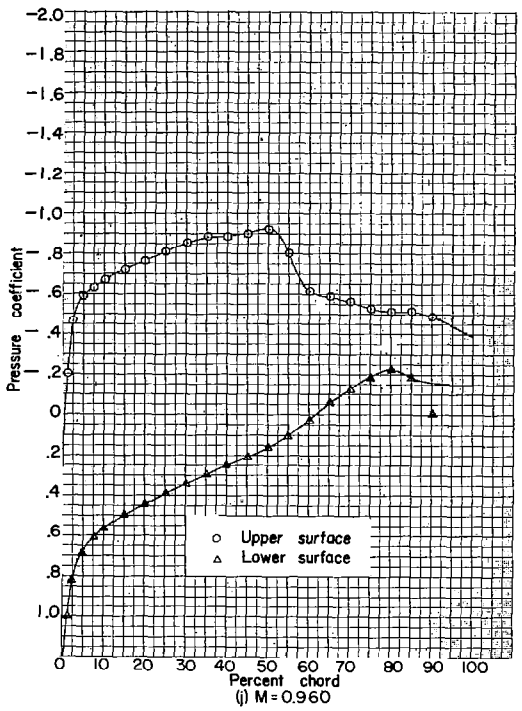
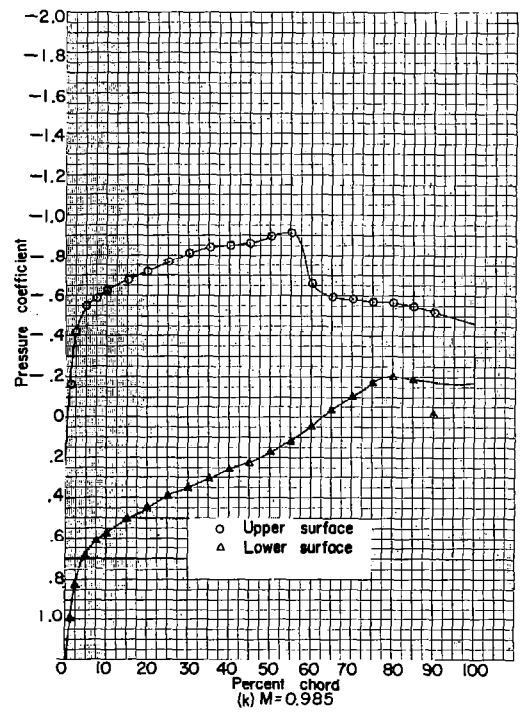


Figure 67.- Continued. NACA 16-512; $\alpha = 10^{\circ}$.







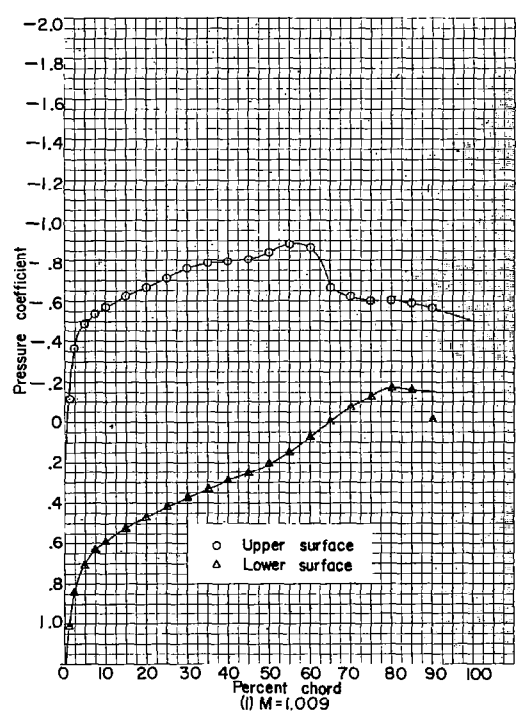
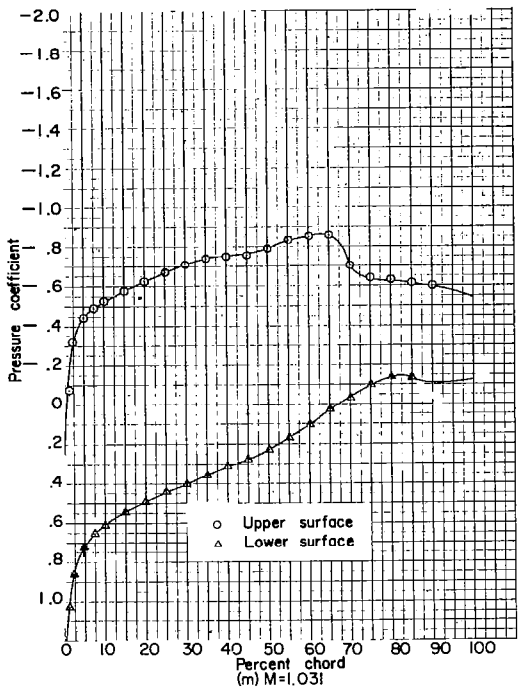
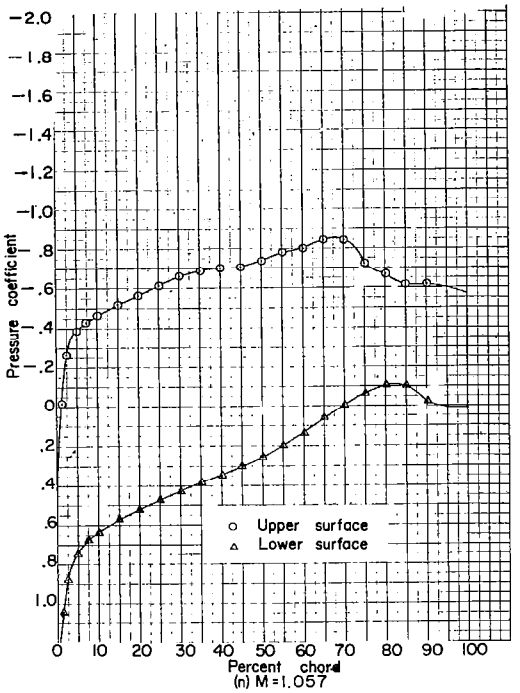
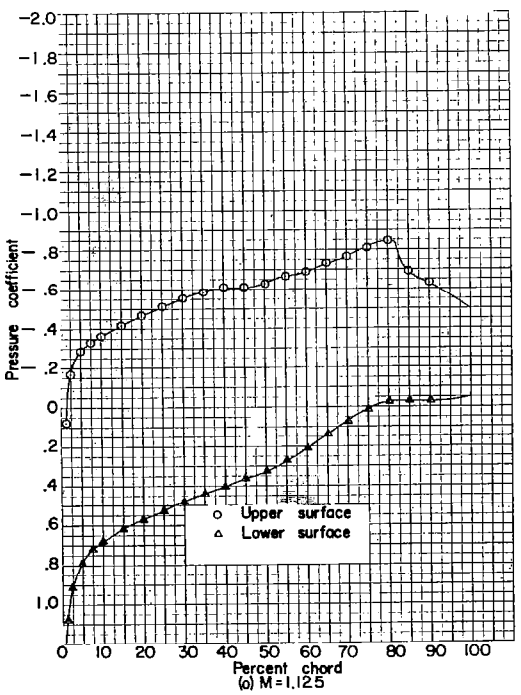


Figure 67.- Continued. NACA 16-512; $\alpha = 10^{\circ}$.







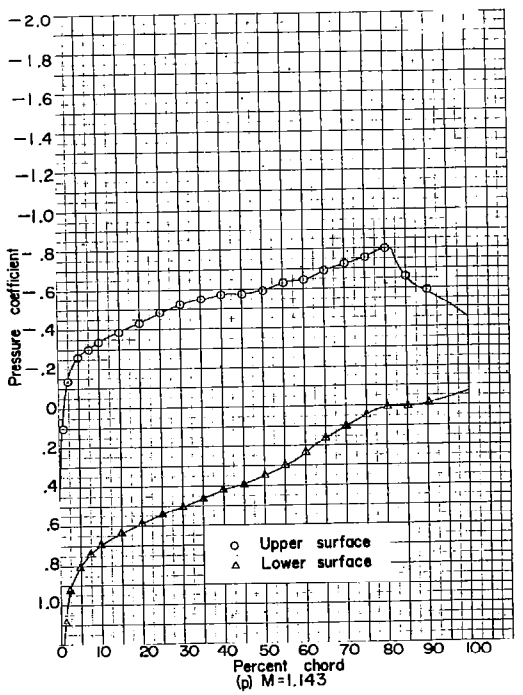


Figure 67.- Concluded. NACA 16-512; $\alpha = 10^{\circ}$.

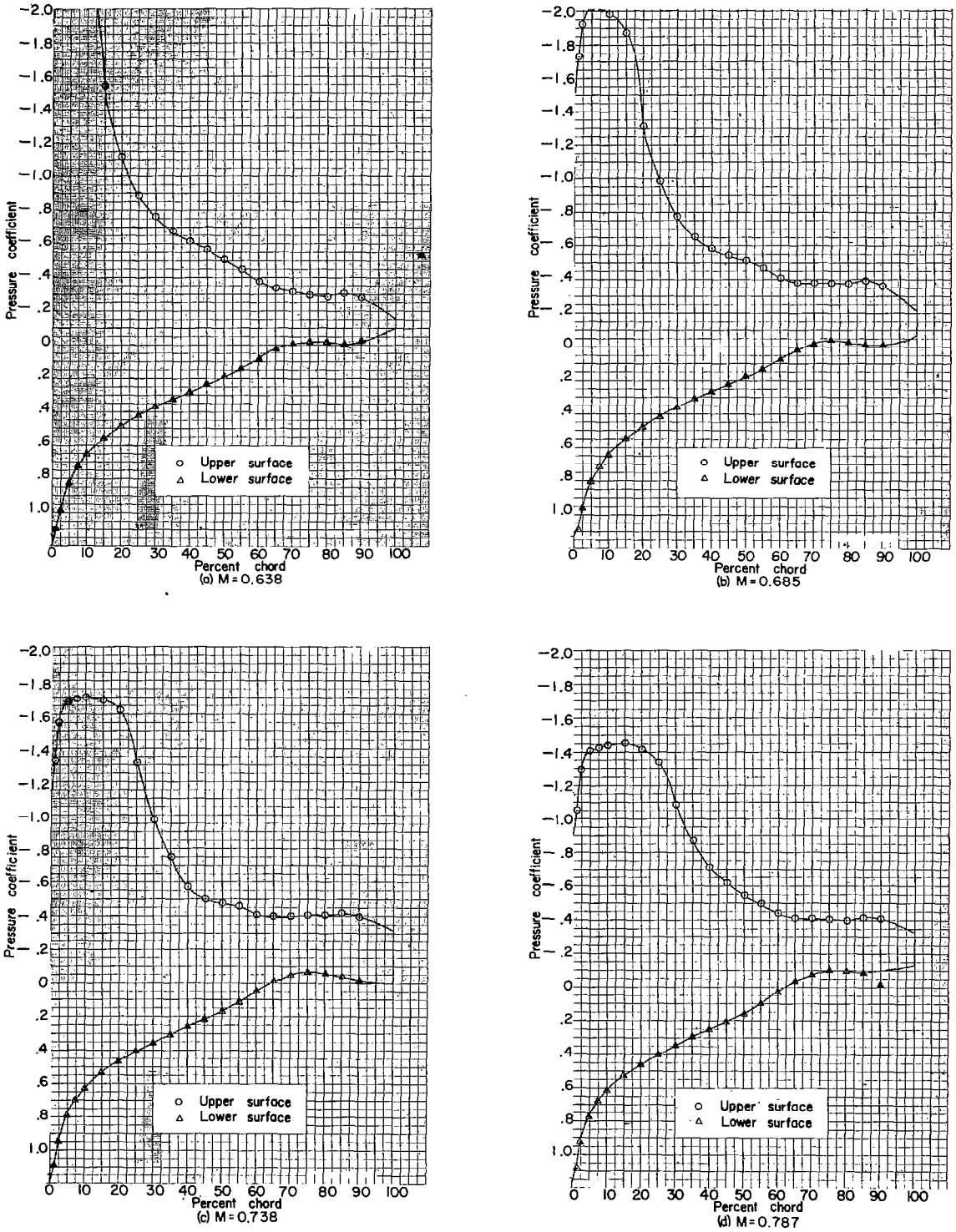
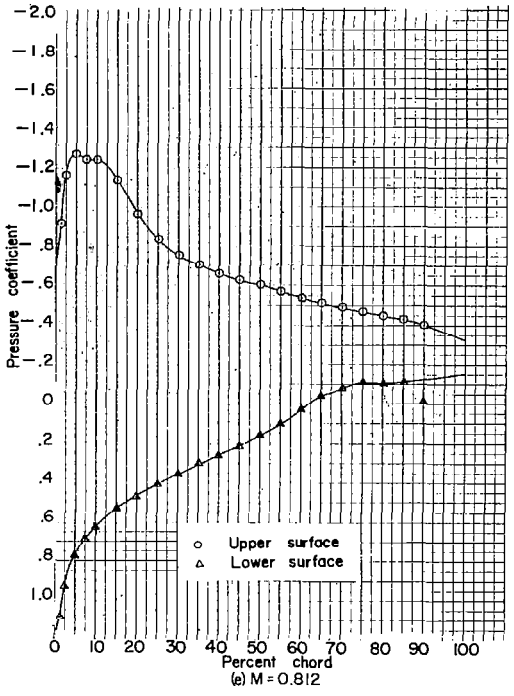
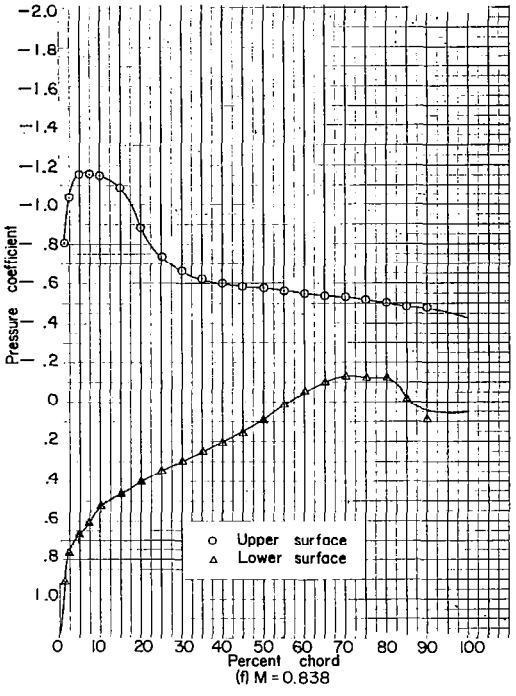
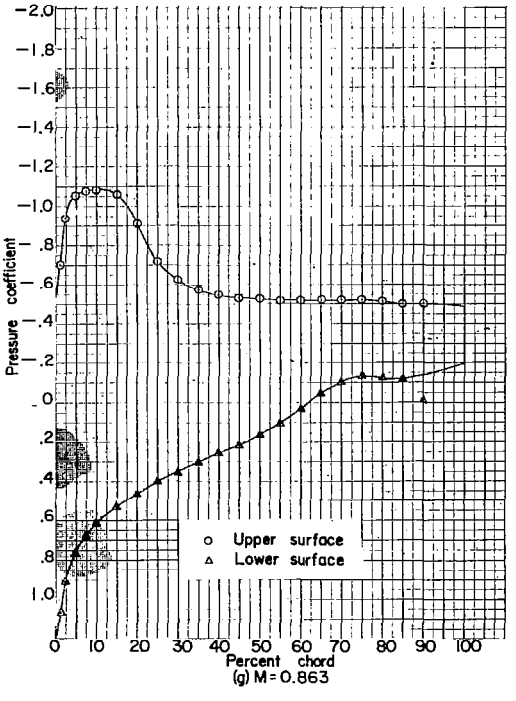


Figure 68.- Pressure distributions over NACA 16-512 airfoil section. $\alpha = 12^{\circ}$.







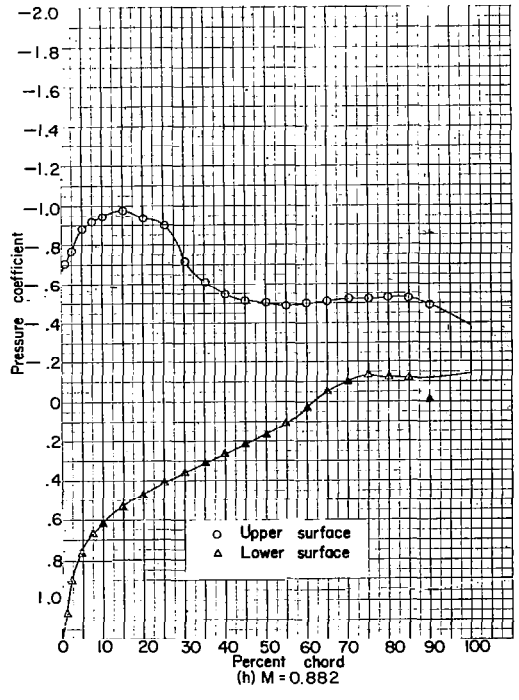
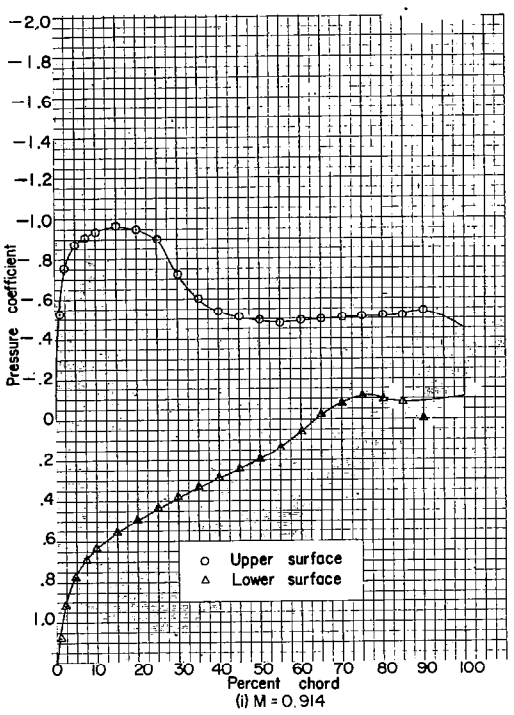
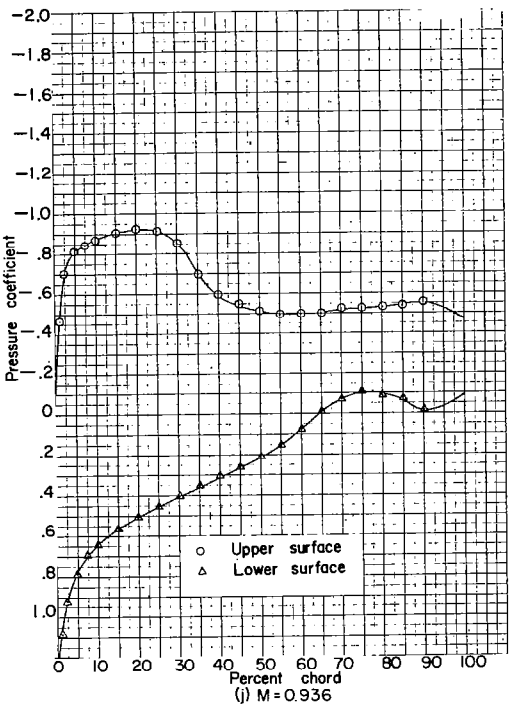
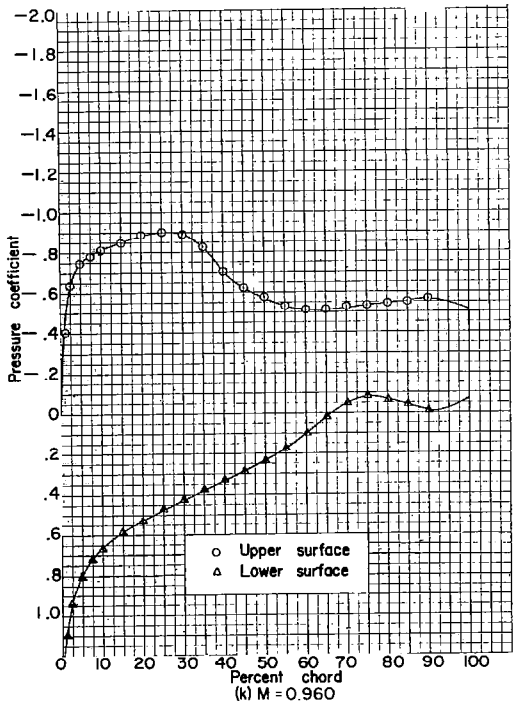


Figure 68.- Continued. NACA 16-512; $\alpha = 12^{\circ}$.







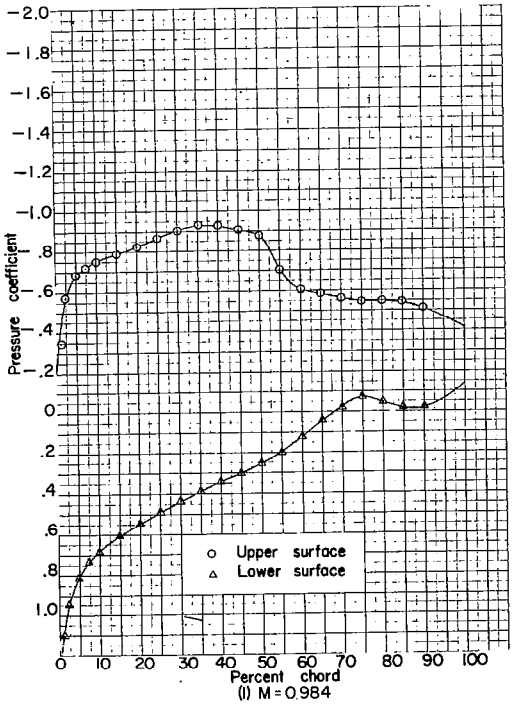


Figure 68.- Continued. NACA 16-512; $\alpha = 12^{\circ}$.

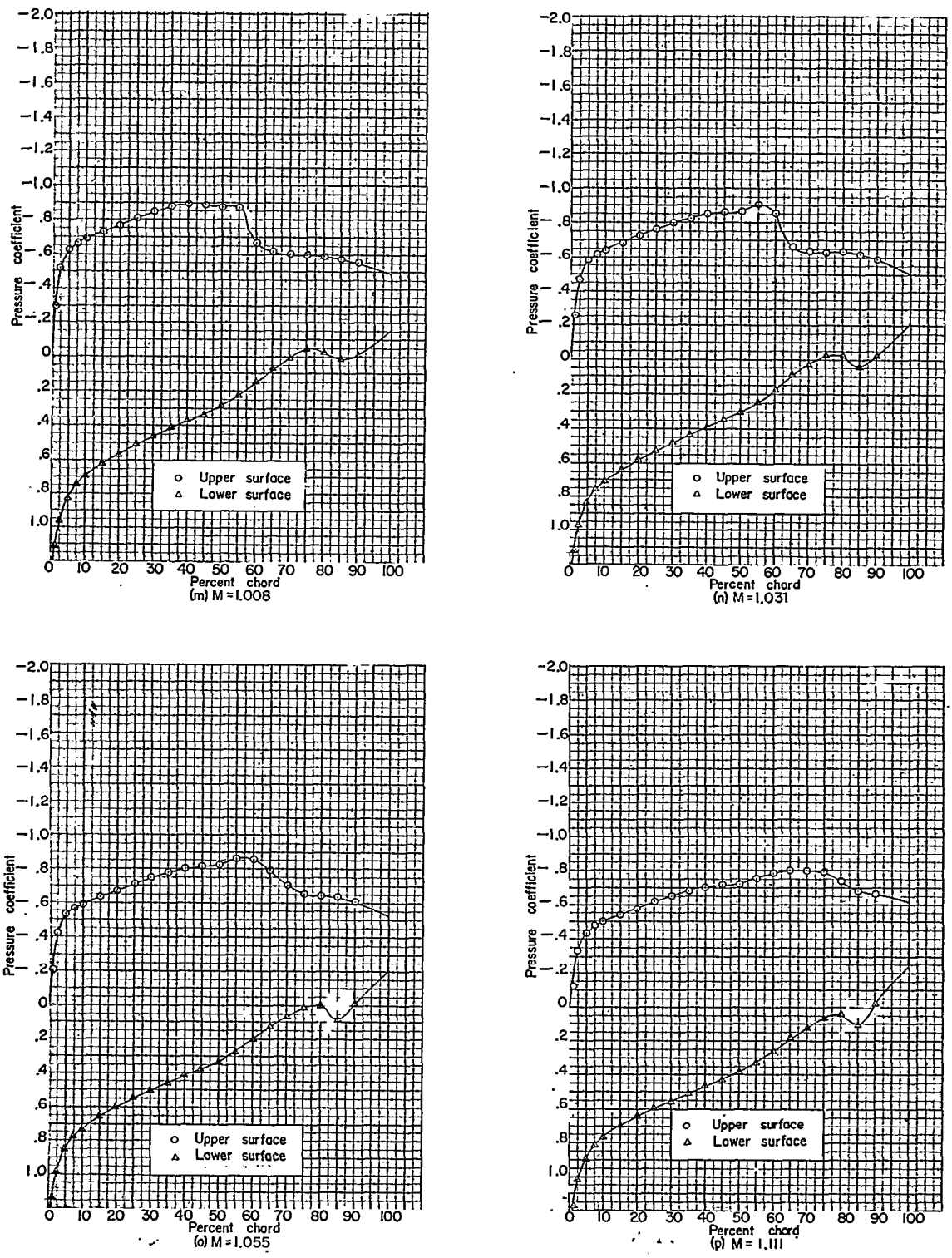


Figure 68.- Concluded. NACA 16-512; $\alpha = 12^{\circ}$.