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TECHNICAL NOTE

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CHORDWISE PRESSURE DISTRIBUTIONS ON A 12-FOOT-SPAN WING OF NACA 66-SERIES AIRFOIL SECTIONS UP TO A MACH NUMBER OF 0.60

By Nancy E. Wall

Langley Aeronautical Laboratory
Langley Field, Va.

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October 1948

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SUMMARY

Chordwise pressure distributions are presented from experimental data obtained at six spanwise stations of a 12-foot-span wing incorporating NACA 66-series sections. The data shown were obtained at Mach numbers ranging from 0.20 to 0.60 and at angles of attack from -4° to the stalling angle of attack.

INTRODUCTION

In view of the rapidly increasing speed and maneuverability requirements of high-speed airplanes, an accurate knowledge of the detailed aerodynamic loads over the wings becomes an essential consideration for an efficiently designed airplane from both structural and aerodynamic aspects. In order to supplement the relatively limited pressure data available on the subject, a comprehensive investigation of the pressures acting over the surfaces of conventional fighter-type wings was undertaken as part of a more complete and extensive test program.

The test wing investigated for the present paper was selected as representative of existing fighter-type airplanes. The analysis of results of the investigation on the present wing is reported in reference 1. The purpose of the present paper is to supplement the information of reference 1 by presenting the basic pressure data obtained over six chordwise stations along the wing semispan for various lift coefficients and Mach numbers.

APPARATUS

These tests were conducted in the Langley 16-foot high-speed tunnel on a test wing having the following geometric properties:

Span, feet	12
Wing area, square feet	24
Aspect ratio	6
Taper ratio	2:1
Wing sections	NACA 66 series (a = 0.6)
Thickness ratio	
Root section, percent	16
Tip section, percent	16
Design lift coefficient	
Root section	0.1
Tip section	0.2
Sweepback (along quarter-chord line), degrees	3.18
Dihedral (along quarter-chord line), degrees	0
Geometric twist (washout), degrees	1.55

The test wing, shown mounted in the tunnel in figure 1 and schematically in figure 2, was machined from solid steel to the section ordinates given in table I. The left semispan of the wing has six chordwise rows of orifices located at 10, 30, 50, 70, 85, and 95 percent of the wing semispan. Each chordwise row contains a leading-edge orifice, 19 upper-surface orifices, and 15 lower-surface orifices. The pressure leads were brought out of the wing from the rear part of the center section by means of a circular pipe fairing attached rigidly to the wing. The leads were then brought through a counterbalanced floating-tail strut to a group of multiple-tube manometers which were photographed during each test.

TESTS AND CORRECTIONS

The pressure tests were conducted over a Mach number range from 0.15 to 0.68 (depending on the model configuration) for a range of angle of attack from -4° to the stalling angle of attack. The data presented herein cover a Mach number range from 0.20 to 0.60. In general, the data for angles of attack below 6° were obtained by maintaining a constant indicated angle of attack and varying the Mach number; whereas for angles of attack above 6°, the tunnel Mach number was held constant and the angle of attack varied to define clearly the stalling characteristics of the wing.

Figure 3 presents the variation of average test Reynolds number with Mach number for the range of these tests. As pointed out in reference 1, the upper limit of this curve corresponds roughly to full-scale performance of current fighter-type airplanes at an altitude of approximately 40,000 feet.

The pressure data presented in this paper, like those of reference 1, are based on the tunnel-empty calibration which was found to yield results in close agreement with the force measurements; no other corrections have been applied to the pressure data. All angle-of-attack corrections were determined during force tests on the wing. The angle of attack was

corrected for air-stream misalinement and jet-boundary effects. The maximum magnitude of the total correction was 1.39°. The corrections are described in detail in reference 1.

PRESENTATION OF RESULTS

Pressures are presented in terms of pressure coefficient P which is defined by the following equation:

$$P = \frac{p - p_o}{q_o}$$

where

p local static pressure, pounds per square foot

p_o free-stream static pressure, pounds per square foot

q_o free-stream dynamic pressure, pounds per square foot

The pressure distributions for each spanwise station at specific Mach numbers and angles of attack are presented in figures 4 to 11. For convenience in locating the data an index of these figures is given in table II.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., April 30, 1948

REFERENCE

1. Cooper, Morton, and Korycinski, Peter F.: The Effects of Compressibility on the Lift, Pressure, and Load Characteristics of a Tapered Wing of NACA 66-Series Airfoil Sections. NACA TN No. 1697, 1948.

TABLE I

AIRFOIL ORDINATES OF 66-SERIES WING

[Stations and ordinates are given in percent of airfoil chord]

Root section				Tip section			
Upper surface		Lower surface		Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate	Station	Ordinate	Station	Ordinate
0	0	0	0	0	0	0	0
.43	1.21	.57	-1.15	.37	1.24	.63	-1.11
.68	1.46	.82	-1.37	.61	1.50	.89	-1.32
1.17	1.82	1.33	-1.68	1.09	1.89	1.41	-1.61
2.41	2.50	2.59	-2.25	2.32	2.61	2.68	-2.13
4.90	3.50	5.10	-3.08	4.79	3.70	5.21	-2.87
7.39	4.28	7.61	-3.73	7.28	4.56	7.72	-3.44
9.89	4.97	10.11	-4.28	9.78	5.31	10.22	-3.93
14.89	6.05	15.11	-5.15	14.79	6.50	15.21	-4.70
19.90	6.89	20.10	-5.83	19.81	7.43	20.19	-5.29
24.92	7.55	25.08	-6.34	24.83	8.16	25.17	-5.74
29.93	8.05	30.07	-6.74	29.86	8.71	30.14	-6.08
34.95	8.41	35.05	-7.02	34.90	9.11	35.10	-6.32
39.97	8.63	40.03	-7.18	39.94	9.36	40.06	-6.46
44.99	8.73	45.01	-7.26	44.98	9.47	45.03	-6.52
50.01	8.69	49.99	-7.22	50.03	9.43	49.98	-6.48
55.04	8.50	54.96	-7.06	55.08	9.23	54.93	-6.34
60.07	8.11	59.93	-6.74	60.14	8.80	59.86	-6.05
65.10	7.46	64.90	-6.20	65.19	8.08	64.81	-5.58
70.10	6.52	69.90	-5.42	70.20	7.07	69.80	-4.86
75.09	5.43	74.91	-4.50	75.18	5.89	74.82	-4.03
80.08	4.23	79.93	-3.49	80.15	4.59	79.85	-3.11
85.05	2.99	84.95	-2.44	85.11	3.26	84.89	-2.17
90.03	1.76	89.97	-1.41	90.06	1.94	89.94	-1.24
95.01	.68	94.99	-.52	95.02	.76	94.98	-.43
100	0	100	0	100	0	100	0
Leading-edge radius = 1.475c Slope of radius through leading edge = 0.058				Leading-edge radius = 1.475c Slope of radius through leading edge = 0.117			

TABLE II.– INDEX OF FIGURES 4 TO 11

M	α (deg)	Figure	M	α (deg)	Figure
0.20	−4.13	4(a)	0.40	−4.13	8(a)
	−1.95	4(b)		−1.95	8(b)
	.26	4(c)		.26	8(c)
	2.42	4(d)		2.42	8(d)
	4.58	4(e)		4.58	8(e)
	6.72	4(f)		6.74	8(f)
	9.92	4(g)		9.98	8(g)
	14.17	4(h)		12.10	8(h)
	15.22	4(i)		14.59	8(i)
	16.27	4(j)		15.22	8(j)
	17.10	4(k)		16.17	8(k)
	17.52	4(l)		17.15	8(l)
	17.52	4(m)			
	18.35	4(n)			
0.25	15.84	5(a)	0.50	6.75	9(a)
	16.68	5(b)		9.98	9(b)
	17.83	5(c)		13.54	9(c)
	18.35	5(d)		14.17	9(d)
	18.35	5(e)		15.19	9(e)
				16.18	9(f)
0.30	6.72	6(a)	0.55	6.81	10(a)
	9.92	6(b)		10.02	10(b)
	14.17	6(c)		12.29	10(c)
	15.22	6(d)		13.31	10(d)
	15.74	6(e)		13.96	10(e)
	16.27	6(f)		15.45	10(f)
	17.30	6(g)			
0.35	6.74	7(a)	0.60	−4.13	11(a)
	9.98	7(b)		−1.95	11(b)
	12.10	7(c)		.26	11(c)
	13.75	7(d)		2.46	11(d)
	14.38	7(e)		4.64	11(e)
	14.59	7(f)		6.81	11(f)
	15.22	7(g)		10.05	11(g)
				11.18	11(h)
				12.82	11(i)
				13.31	11(j)
				13.60	11(k)

NACA

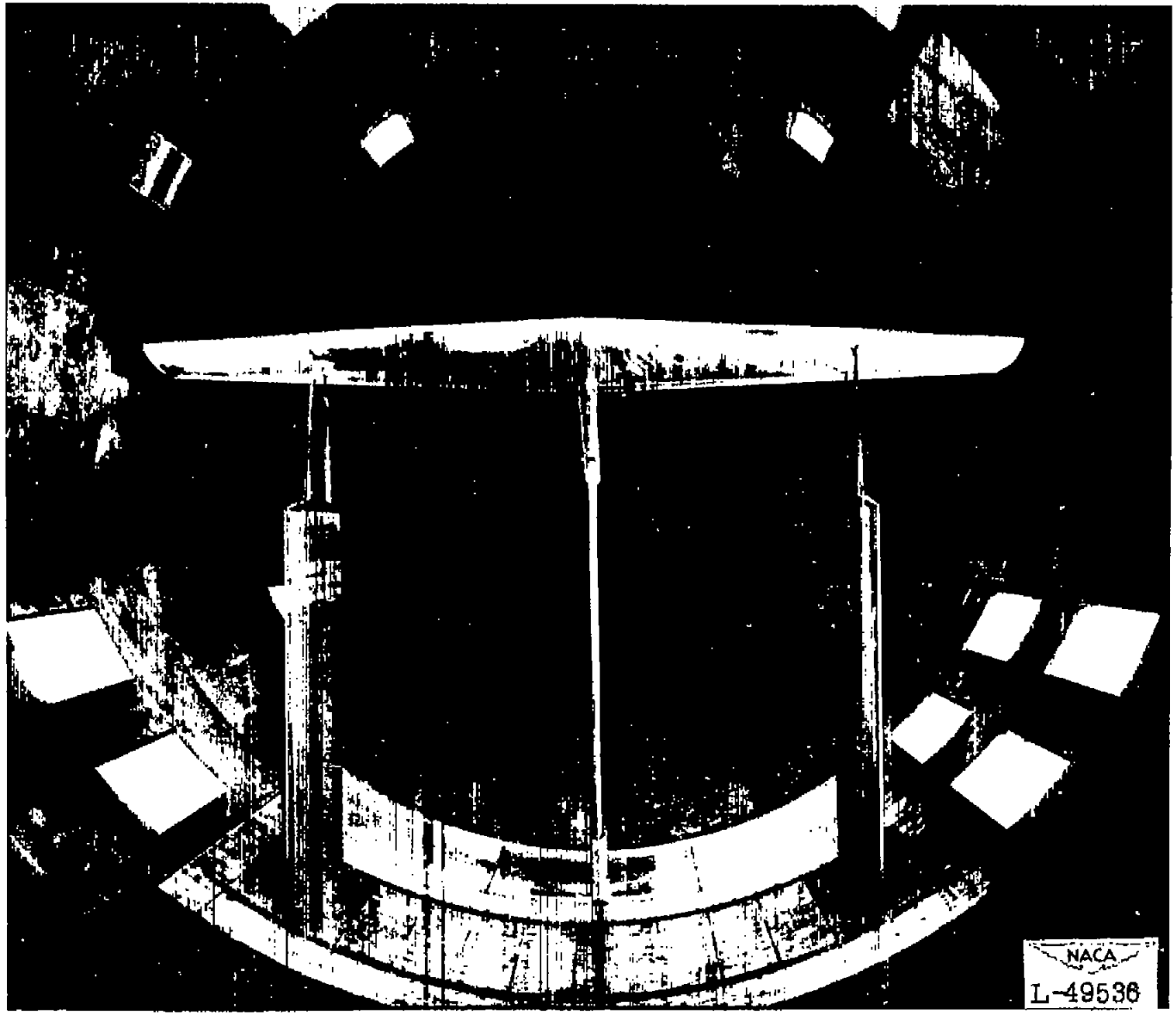


Figure 1.- Downstream view of test wing mounted for pressure tests.

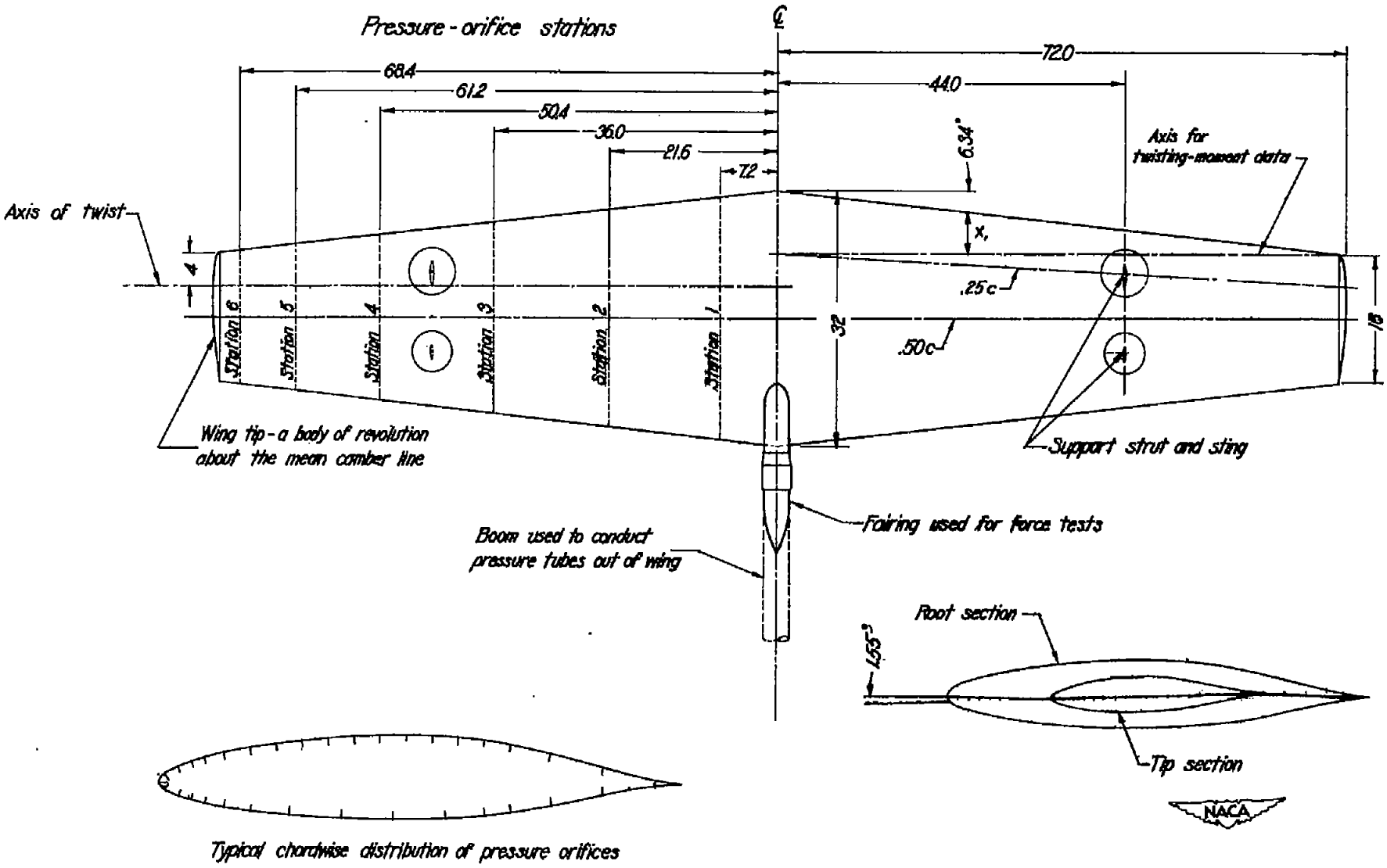


Figure 2.- Principal wing dimensions and locations of pressure orifices. (All dimensions are in inches.)

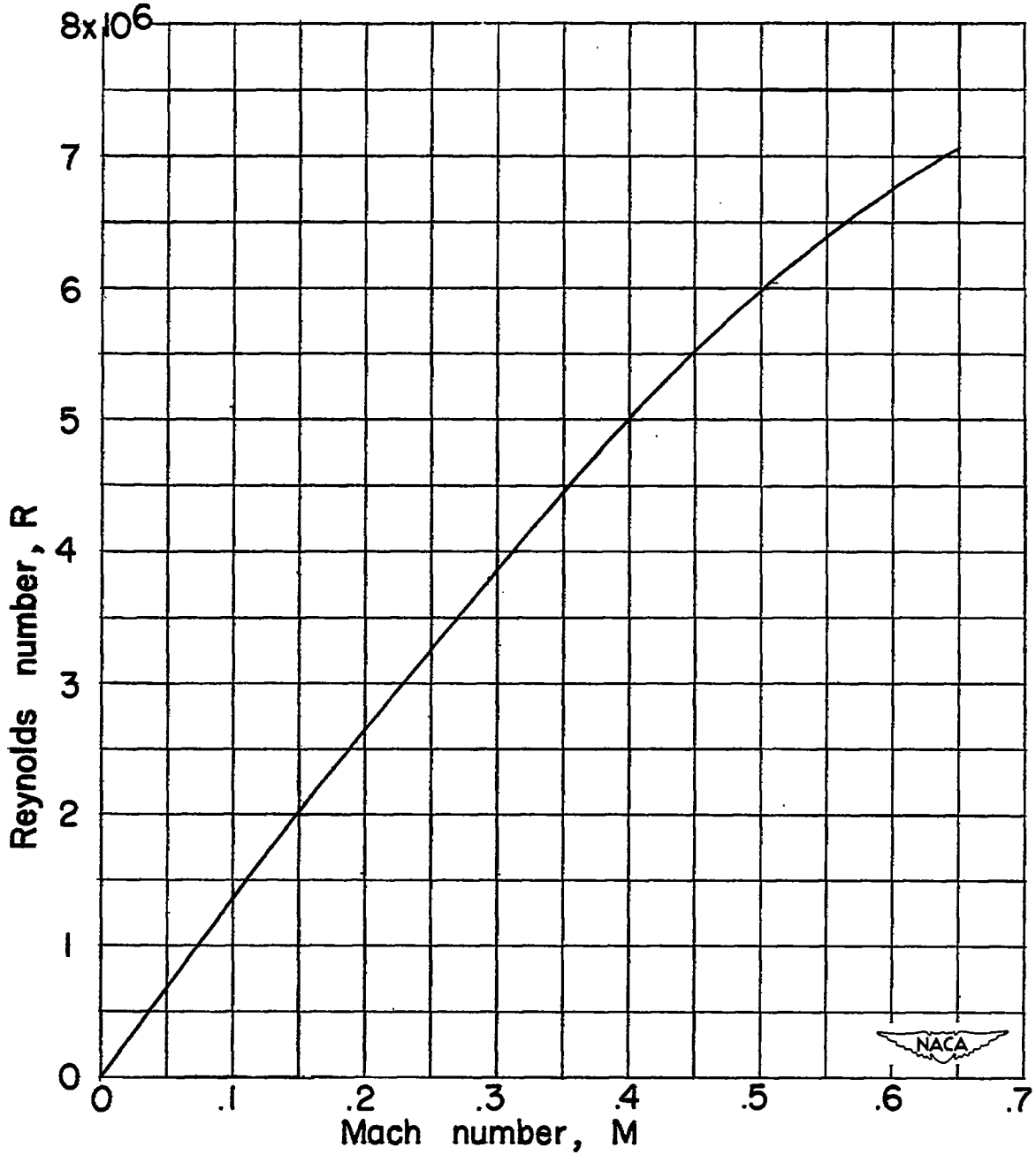
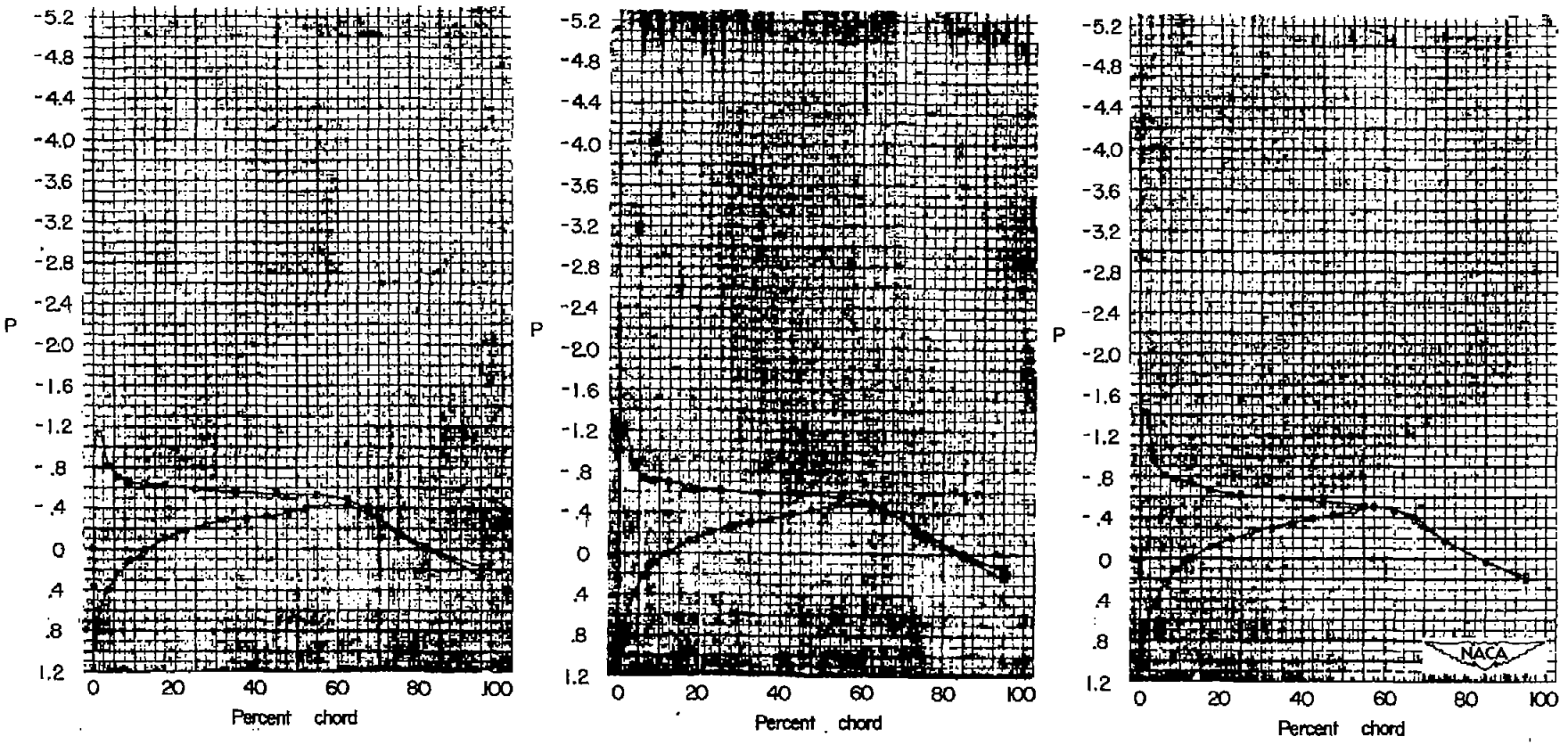


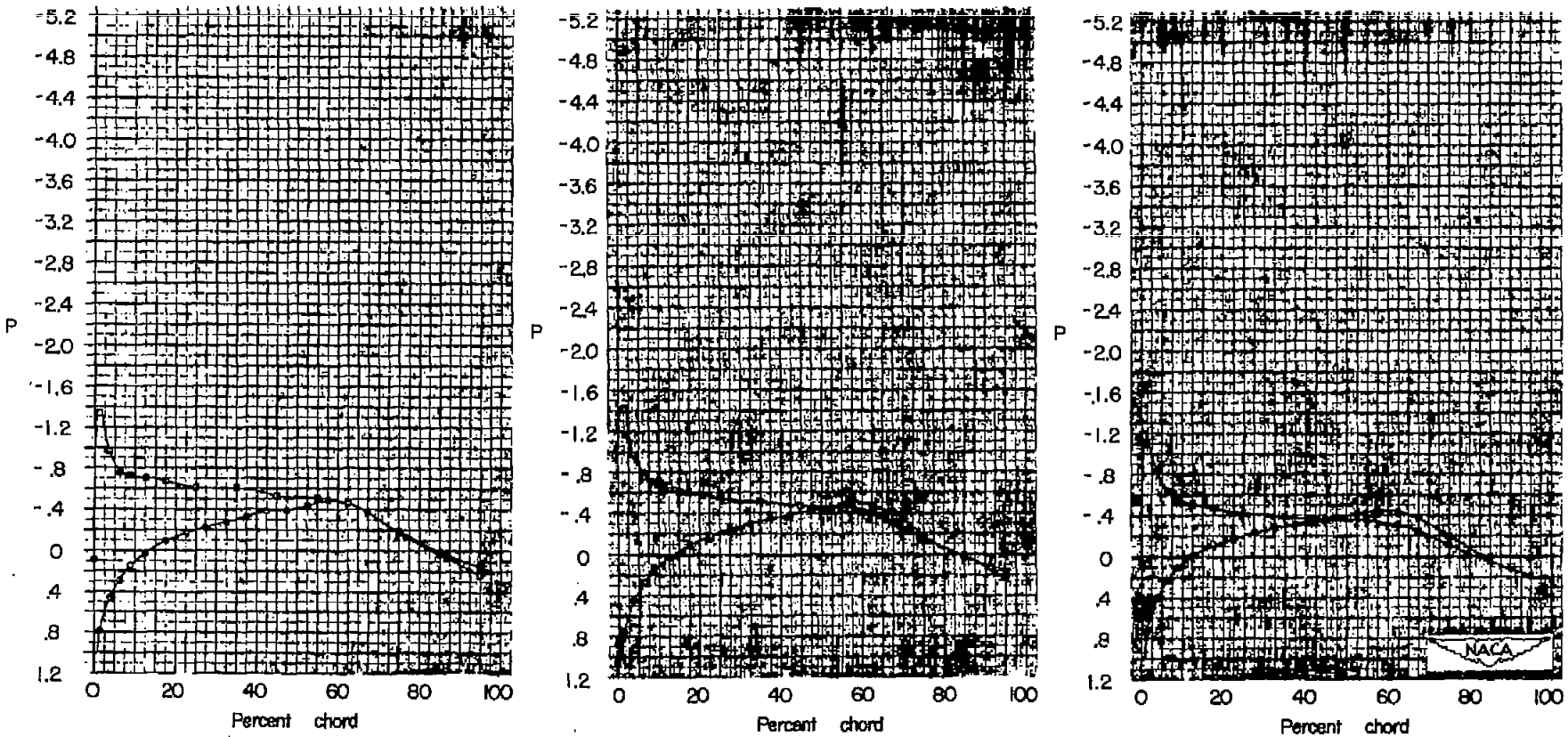
Figure 3.- Variation of average test Reynolds number with Mach number.



Station 1 Station 2 Station 3

(a) $\alpha = -4.13^{\circ}$.

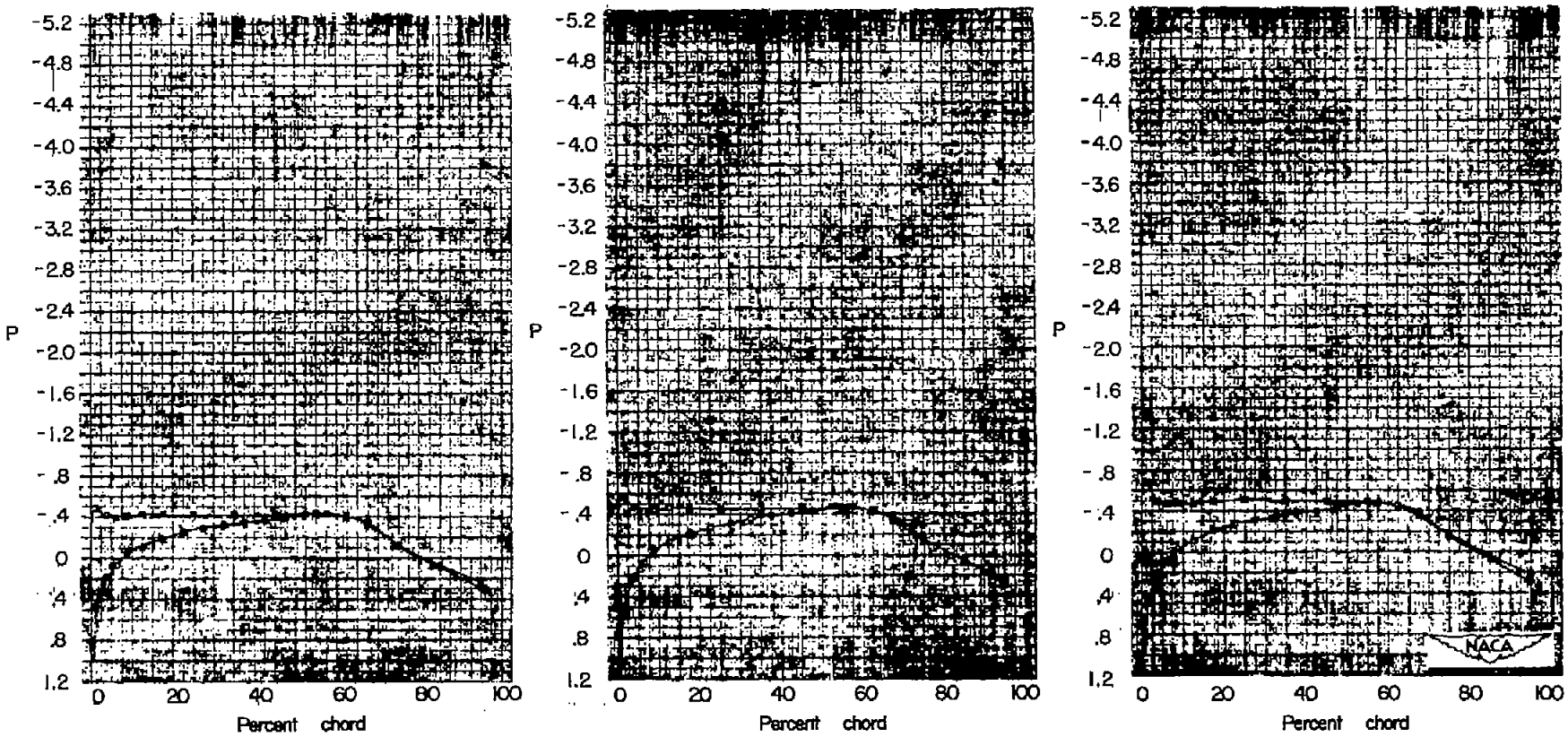
Figure 4.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.20.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = -4.13^{\circ}$.

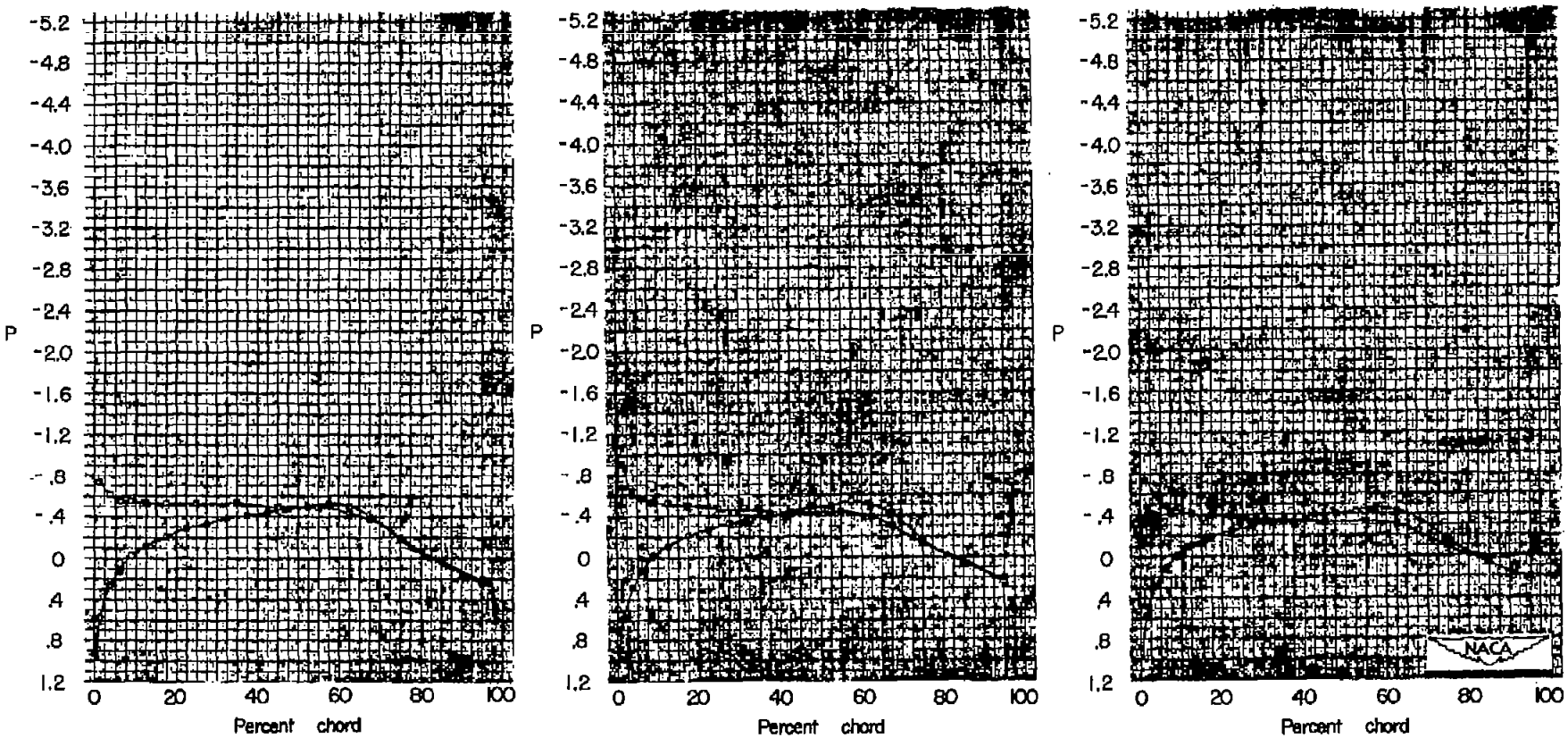
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(b) $\alpha = -1.95^{\circ}$.

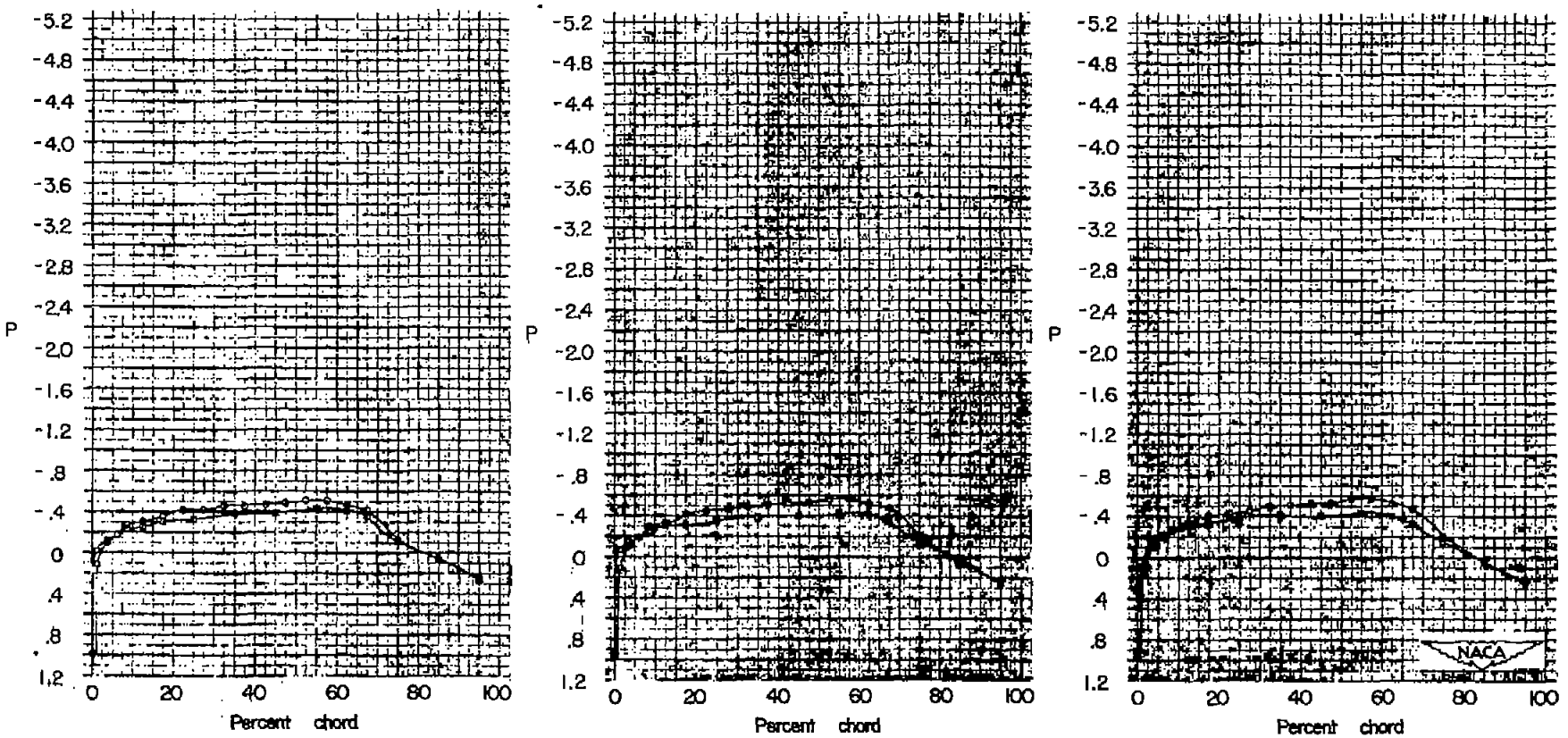
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = -1.95^{\circ}$.

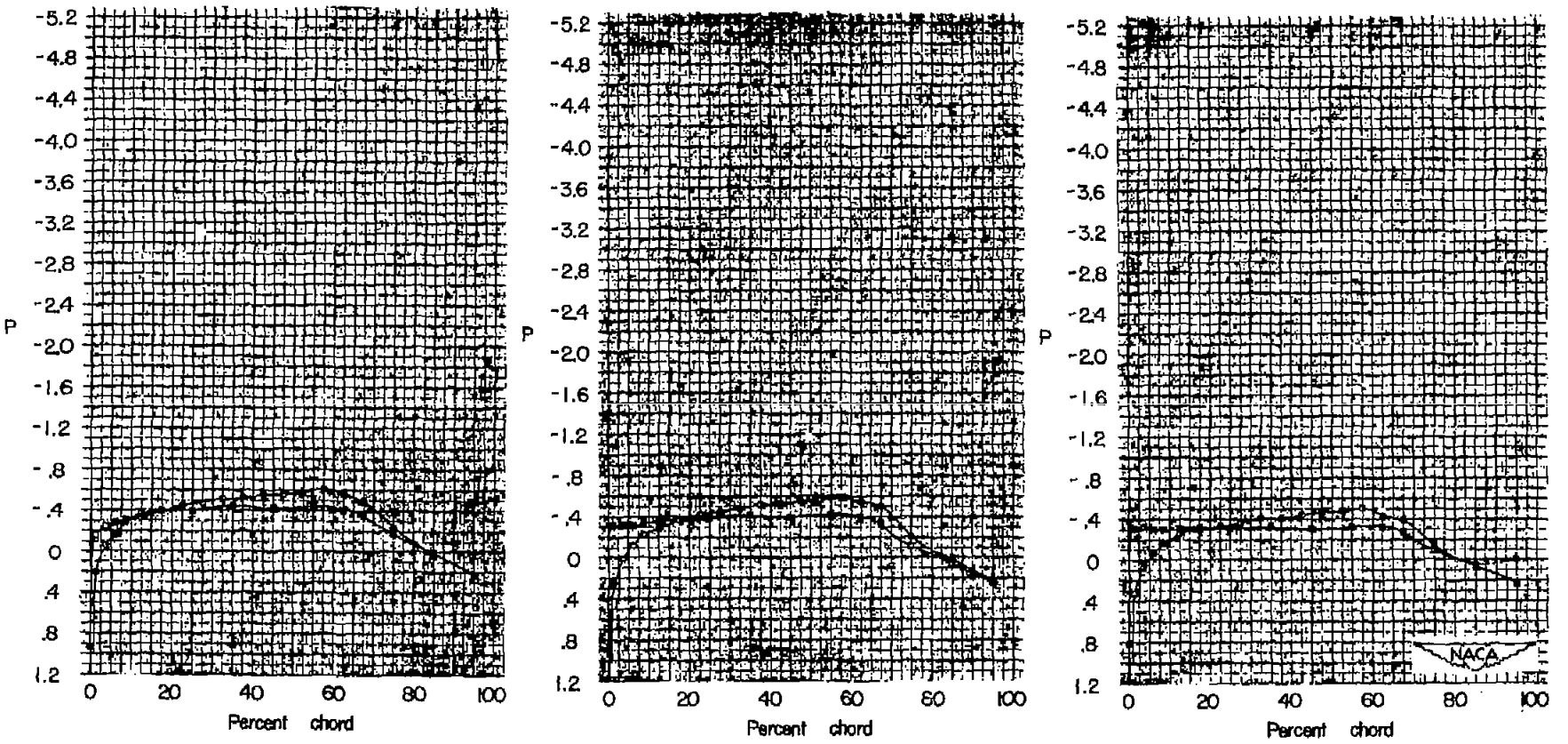
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(c) $\alpha = 0.26^{\circ}$.

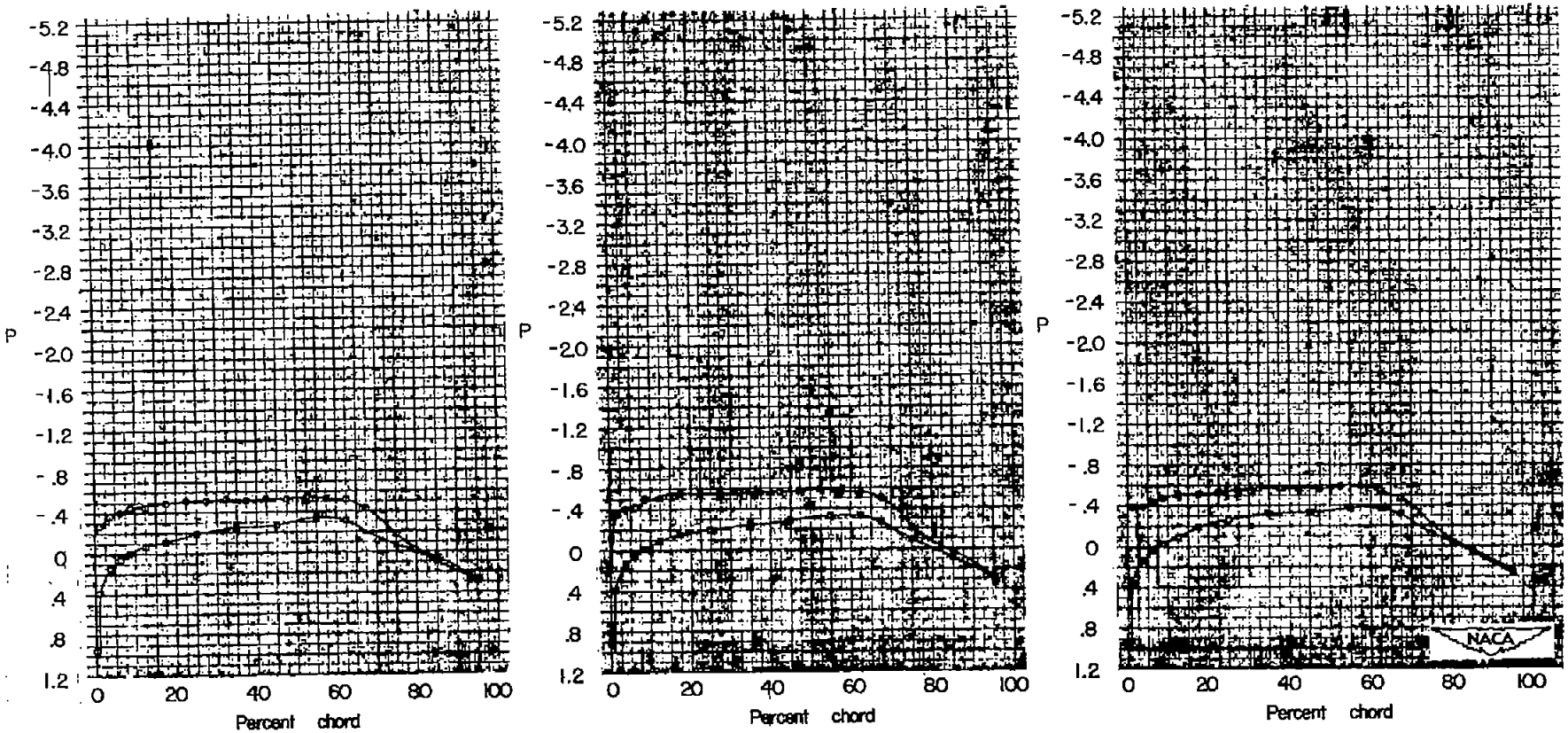
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 0.26^{\circ}$.

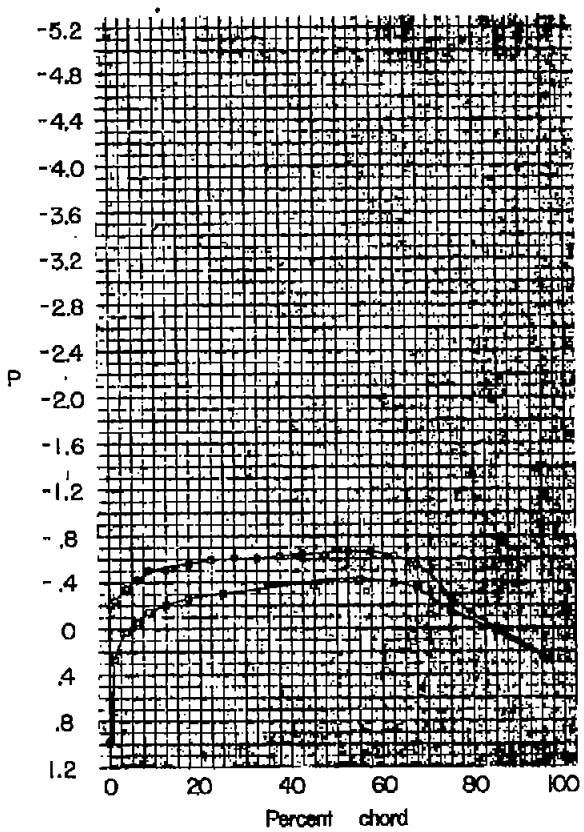
Figure 4.- Continued. M = 0.20.



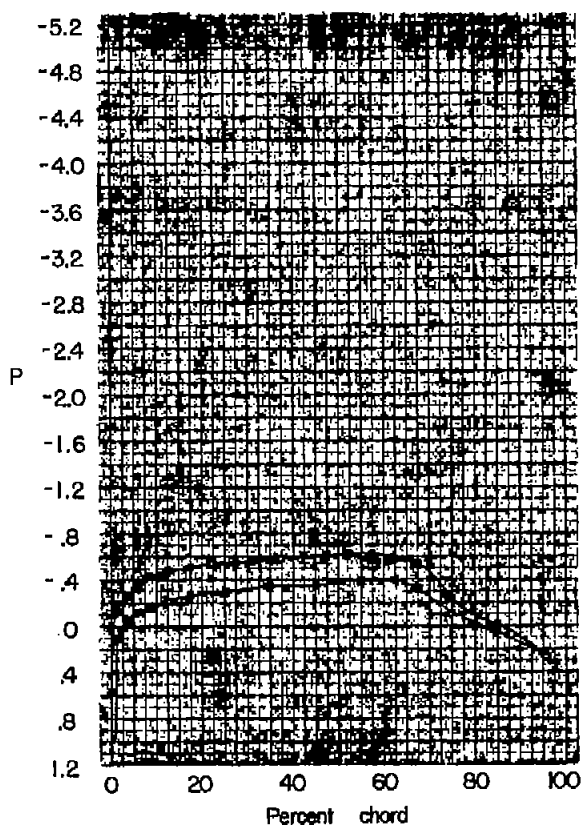
Station 1 Station 2 Station 3

(d) $\alpha = 2.42^{\circ}$.

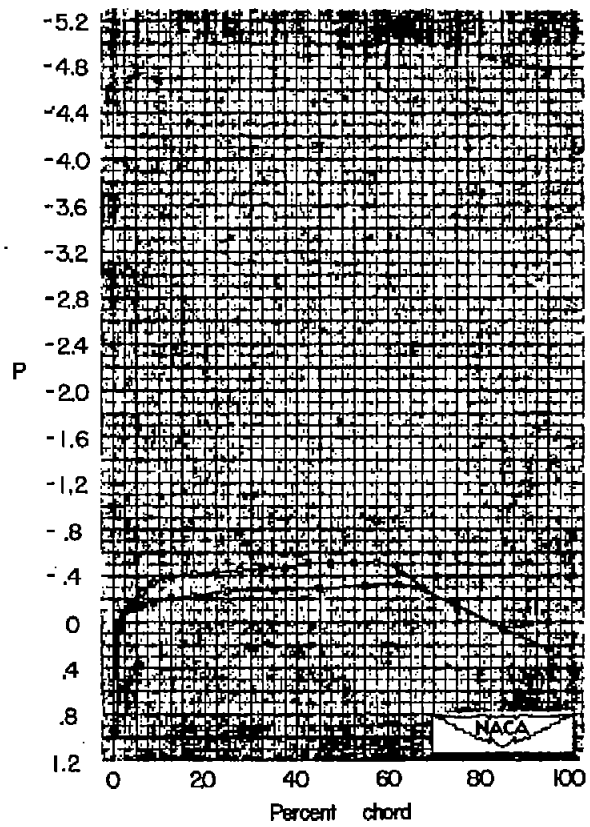
Figure 4.- Continued. M = 0.20.



Station 4



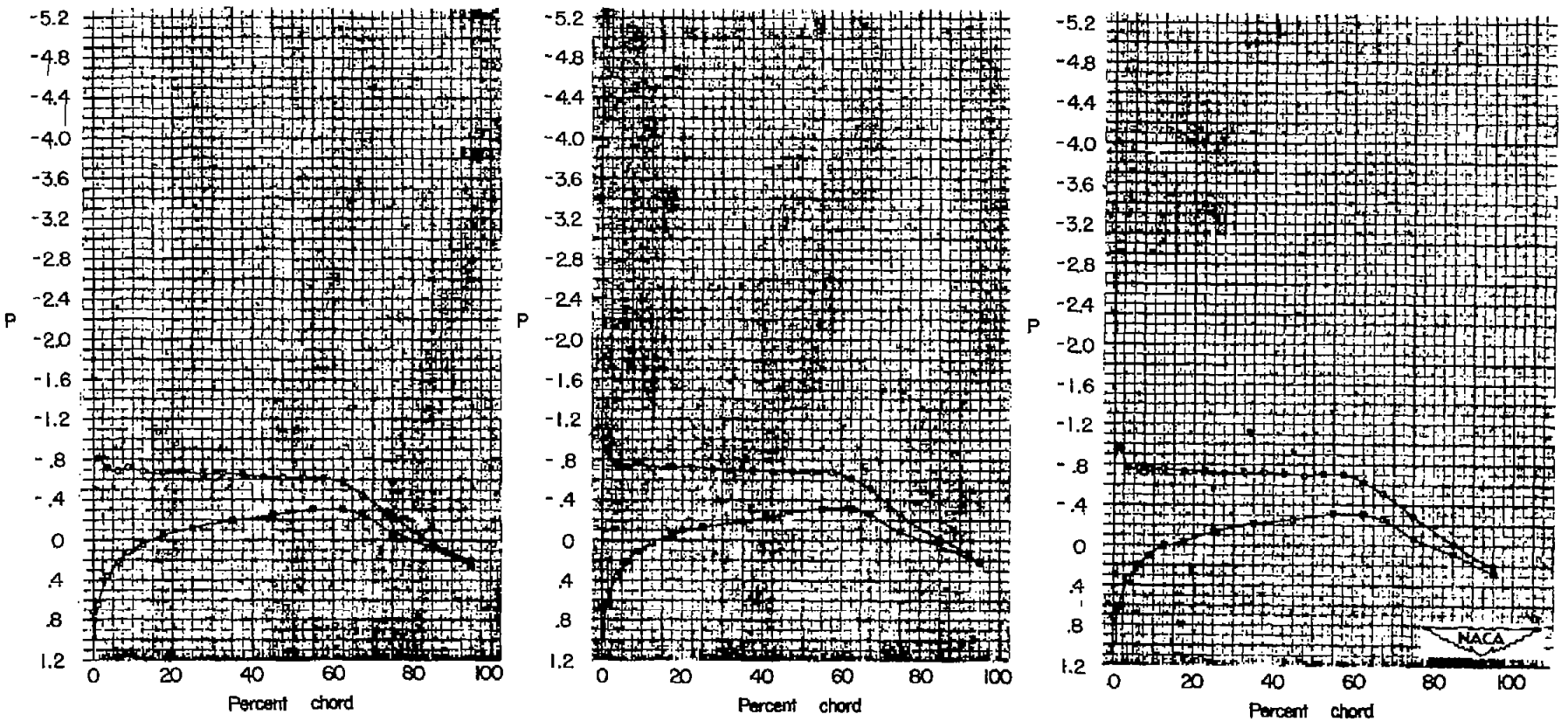
Station 5



Station 6

(d) Concluded. $\alpha = 2.42^{\circ}$.

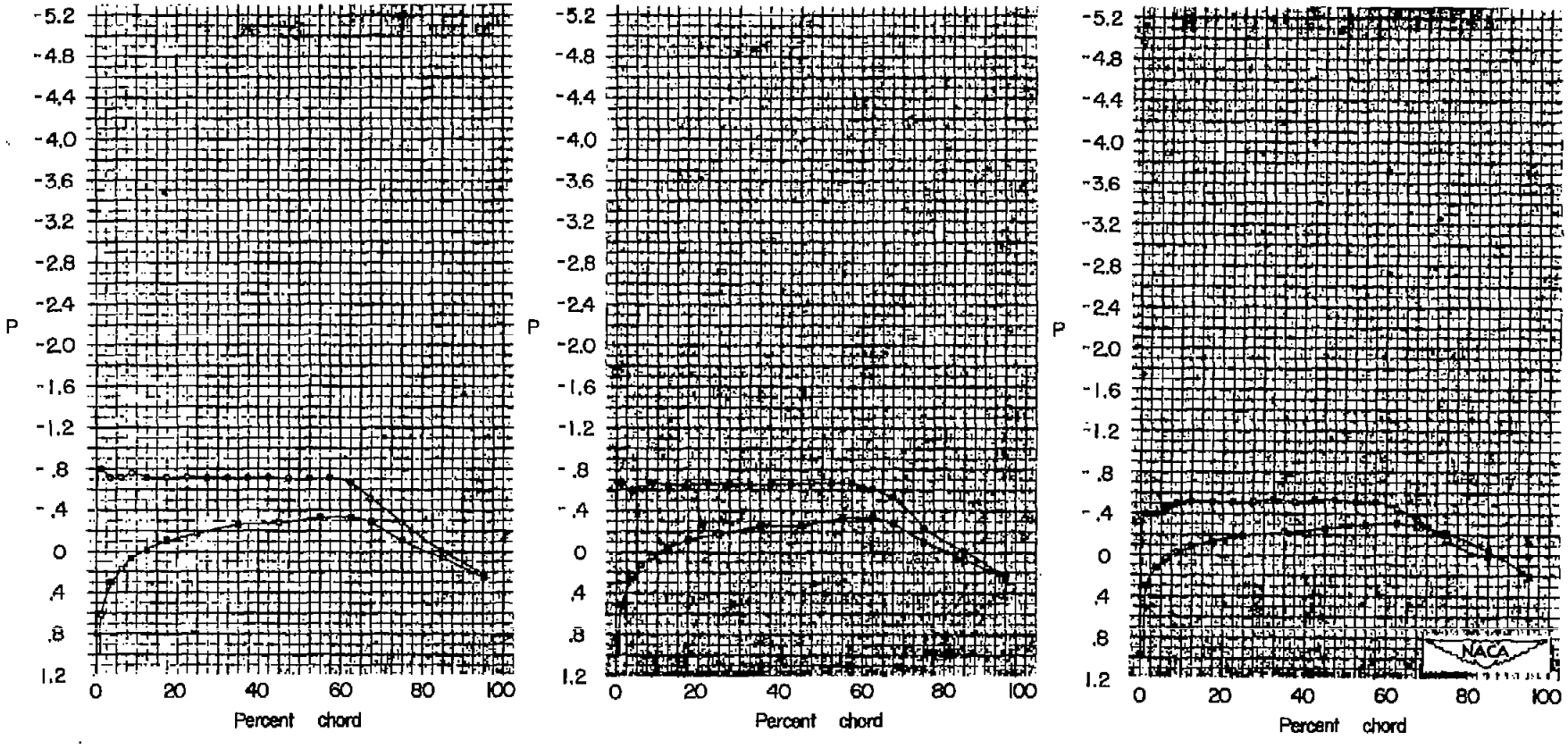
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(e) $\alpha = 4.58^{\circ}$.

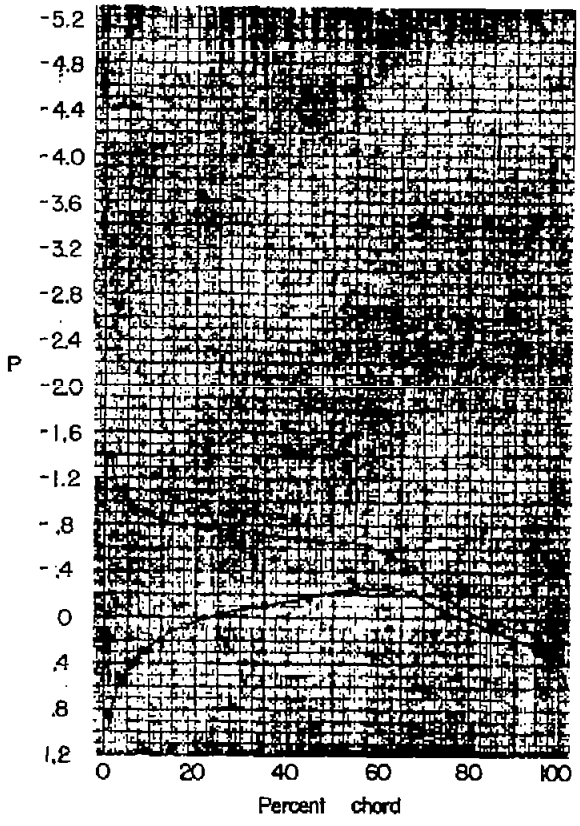
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 4.58^{\circ}$.

Figure 4.- Continued. M = 0.20.



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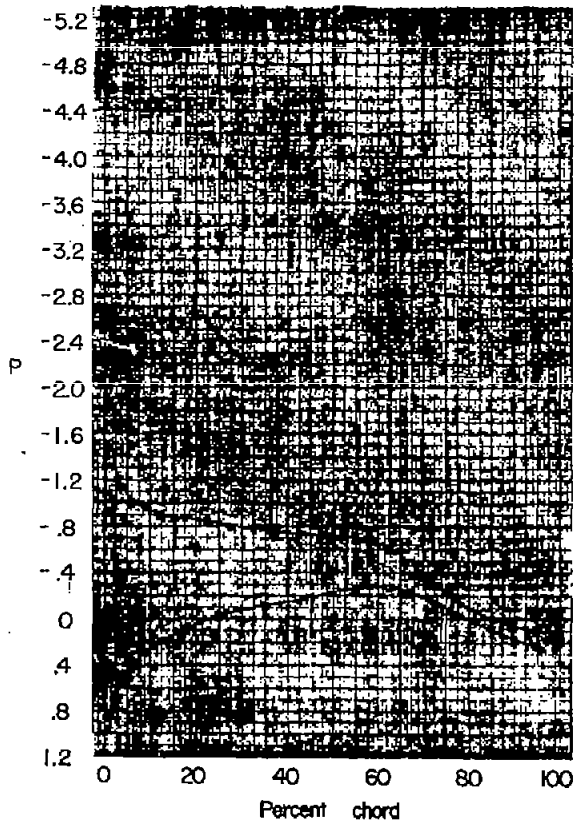
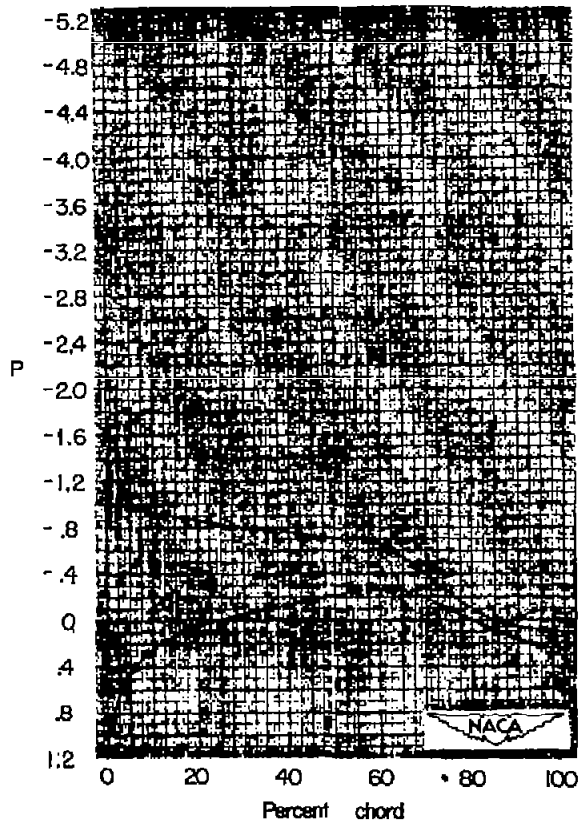


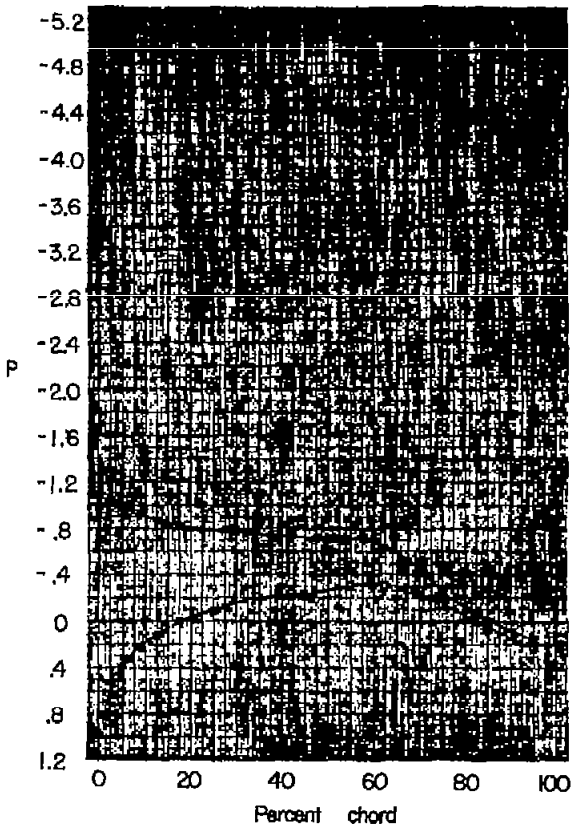
Station 2



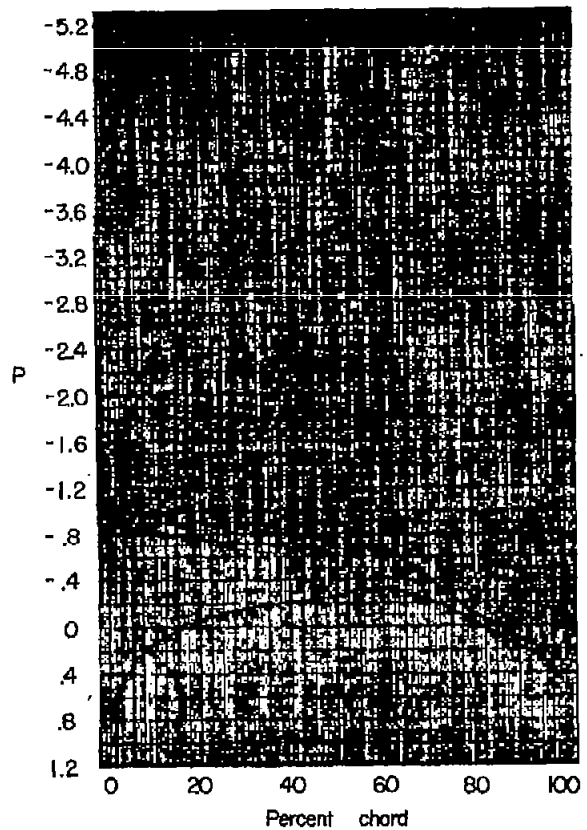
Station 3

(f) $\alpha = 6.72^{\circ}$.

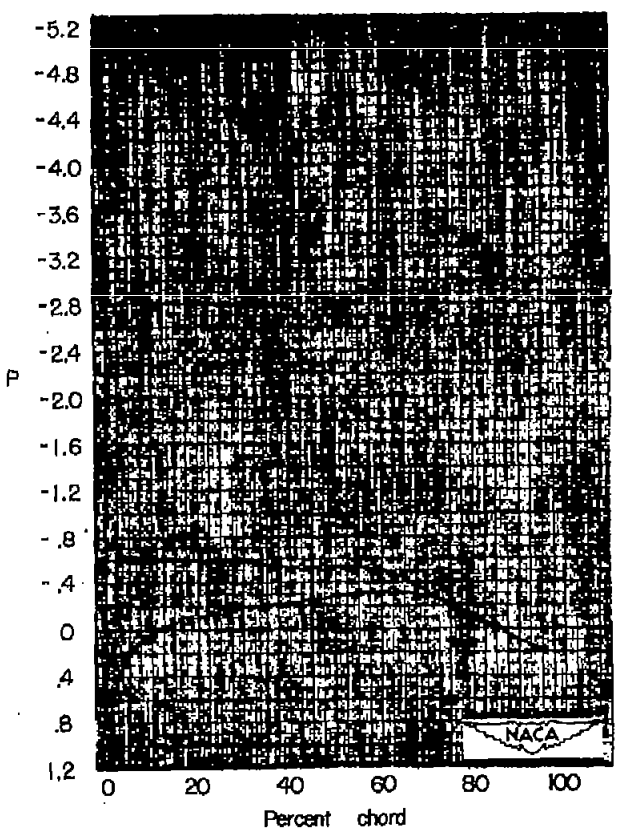
Figure 4.- Continued. M = 0.20.



Station 4



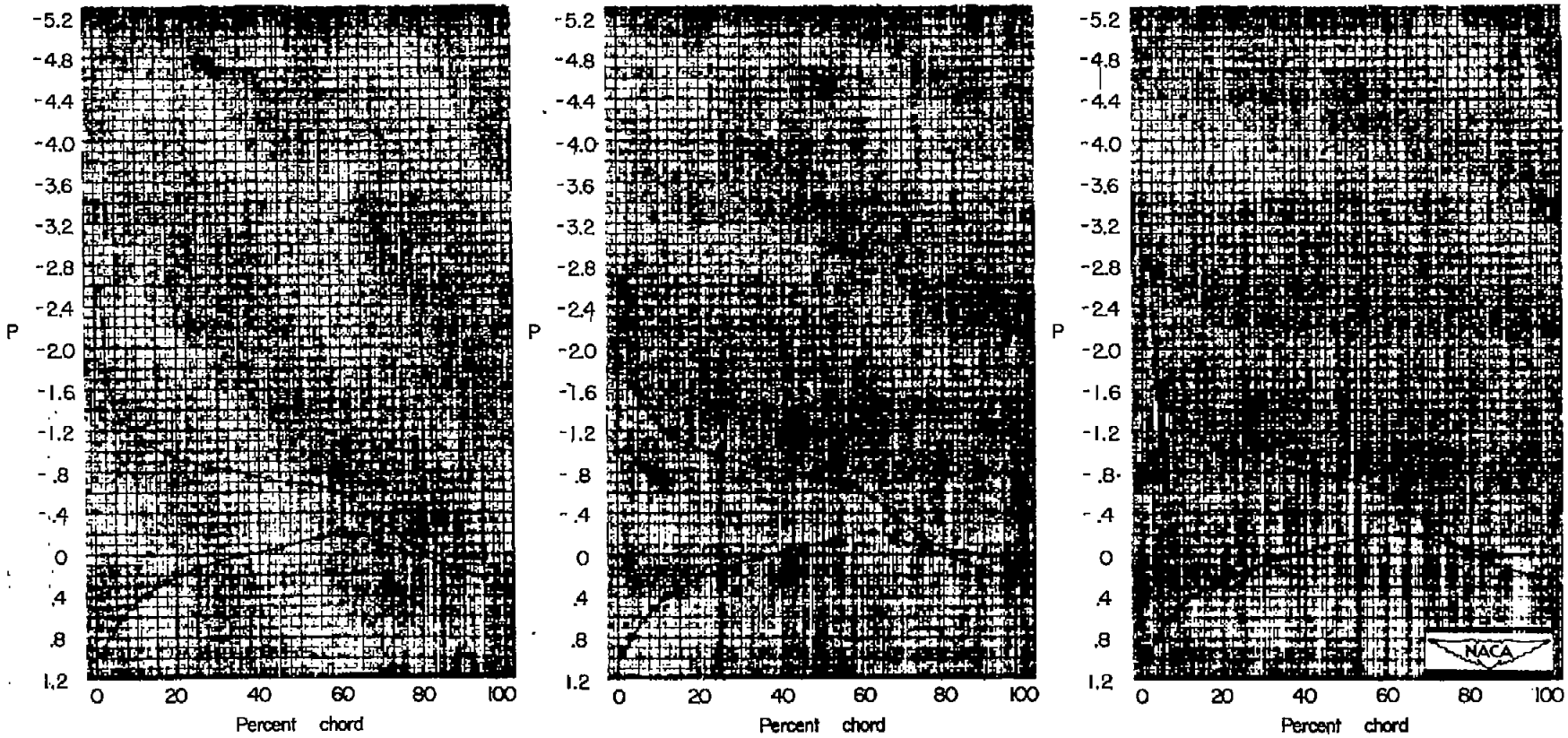
Station 5



Station 6

(f) Concluded. $\alpha = 6.72^{\circ}$.

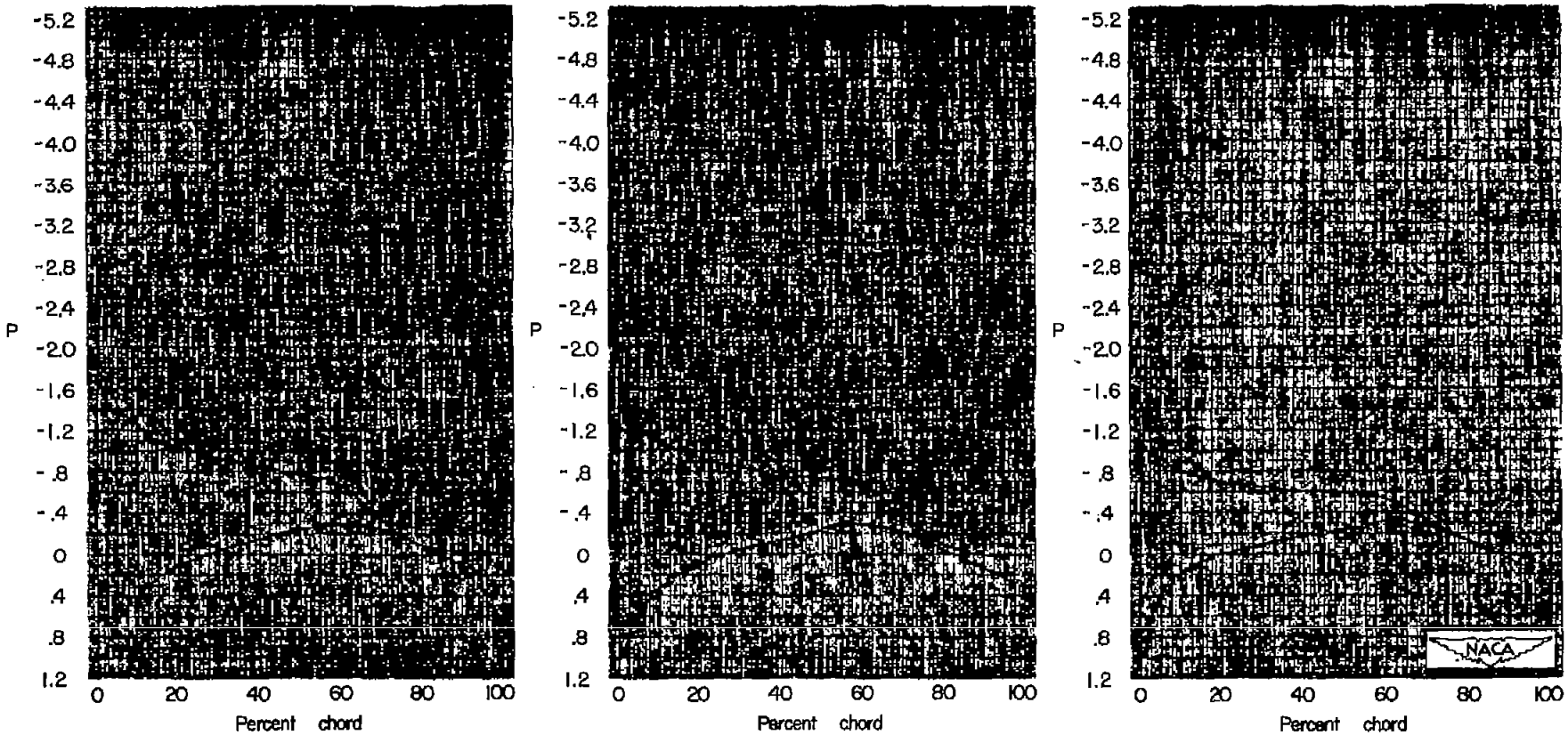
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(g) $\alpha = 9.92^{\circ}$.

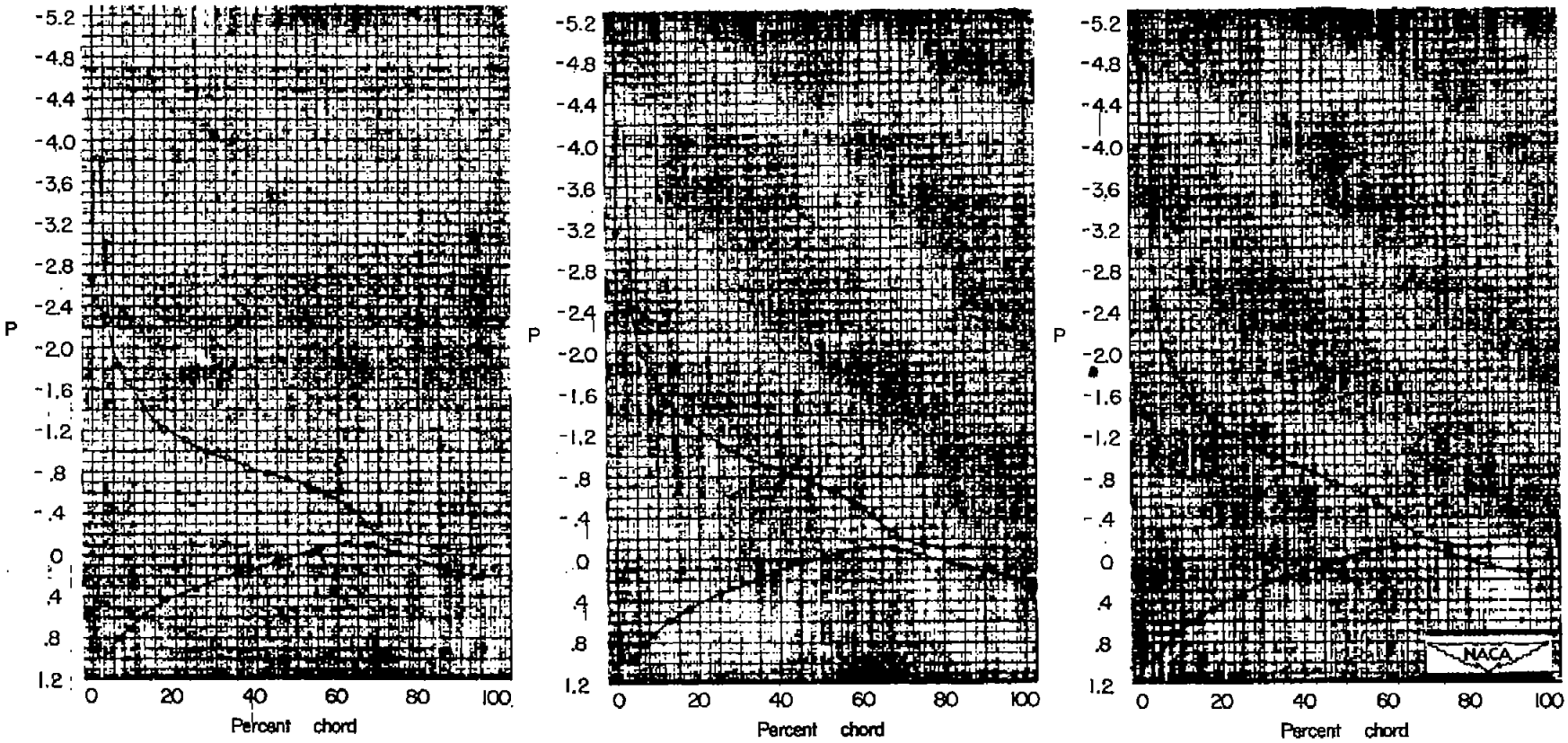
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(g) Concluded. $\alpha = 9.92^{\circ}$.

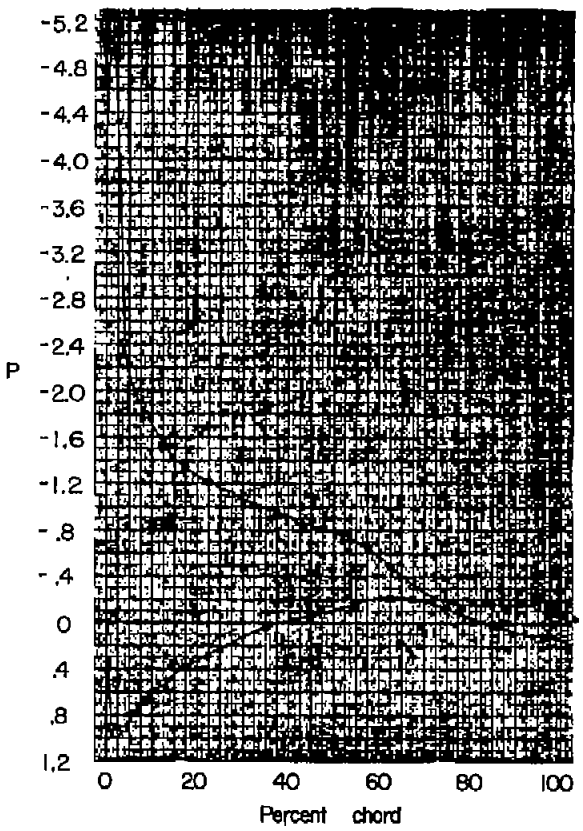
Figure 4.- Continued. M = 0.20.



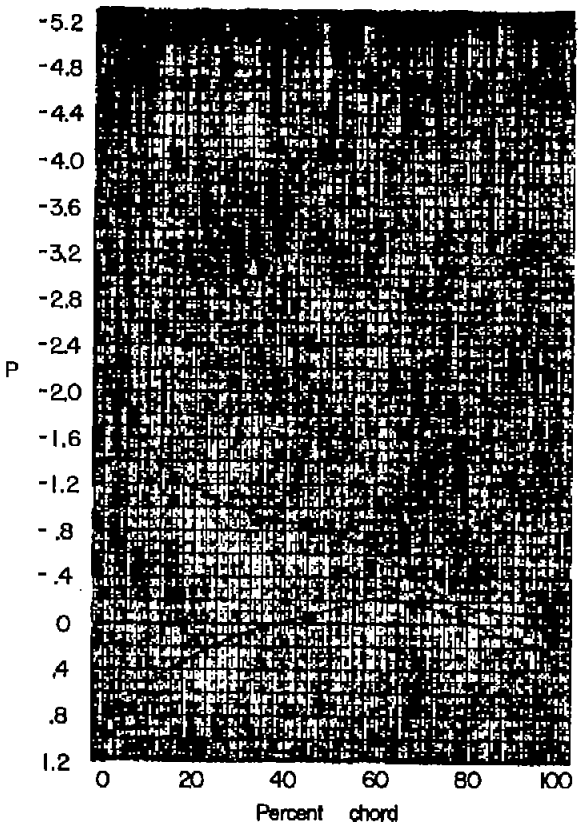
Station 1 Station 2 Station 3

(h) $\alpha = 14.17^{\circ}$.

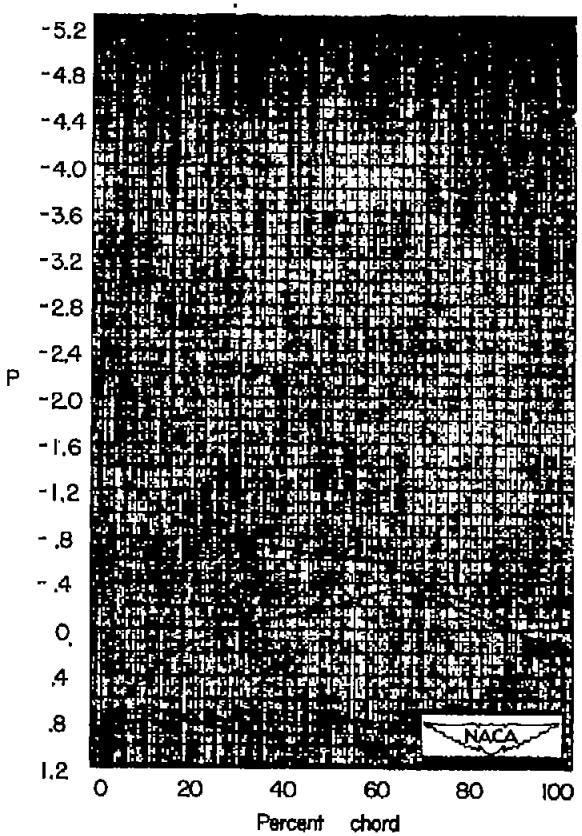
Figure 4.- Continued. M = 0.20.



Station 4



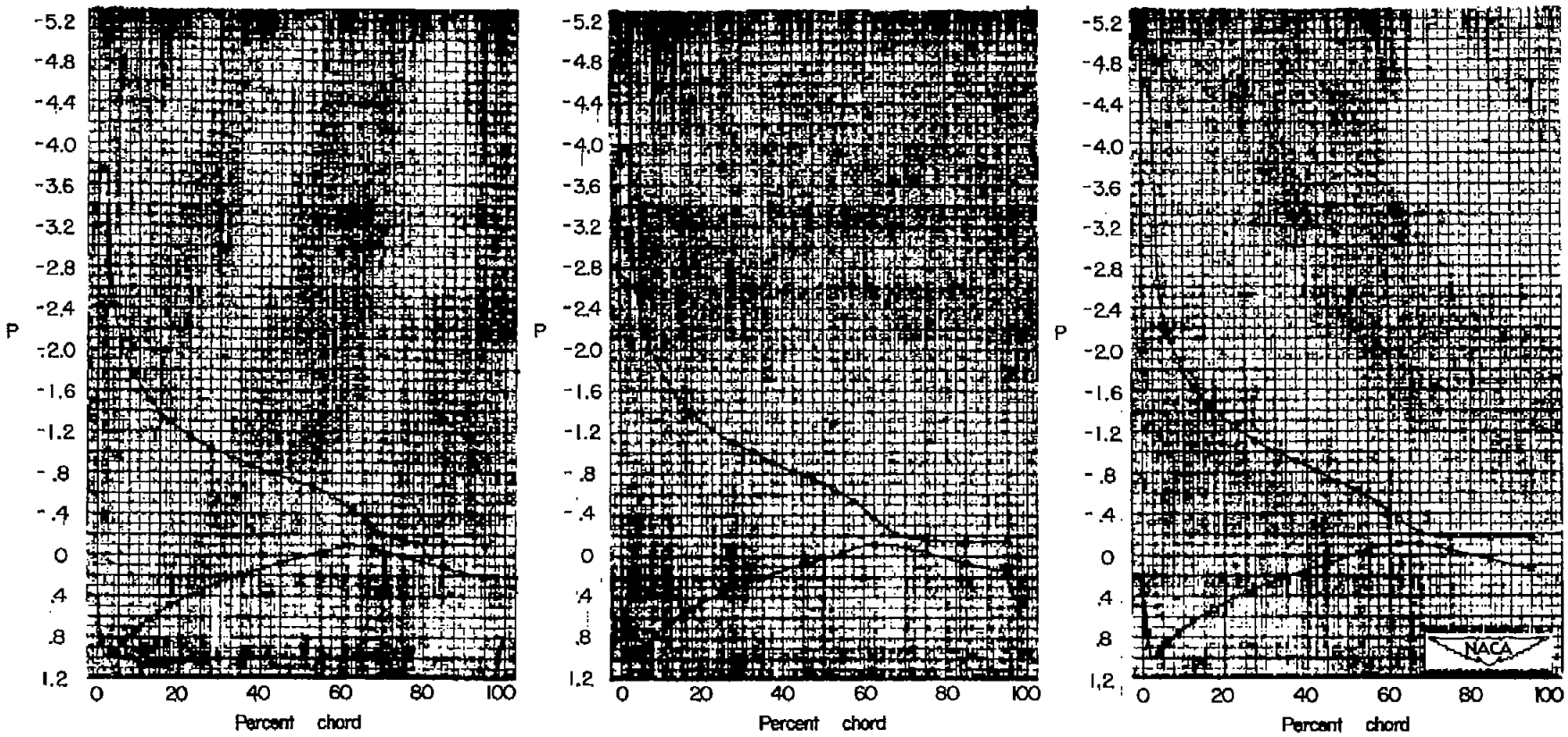
Station 5



Station 6

(h) Concluded. $\alpha = 14.17^{\circ}$.

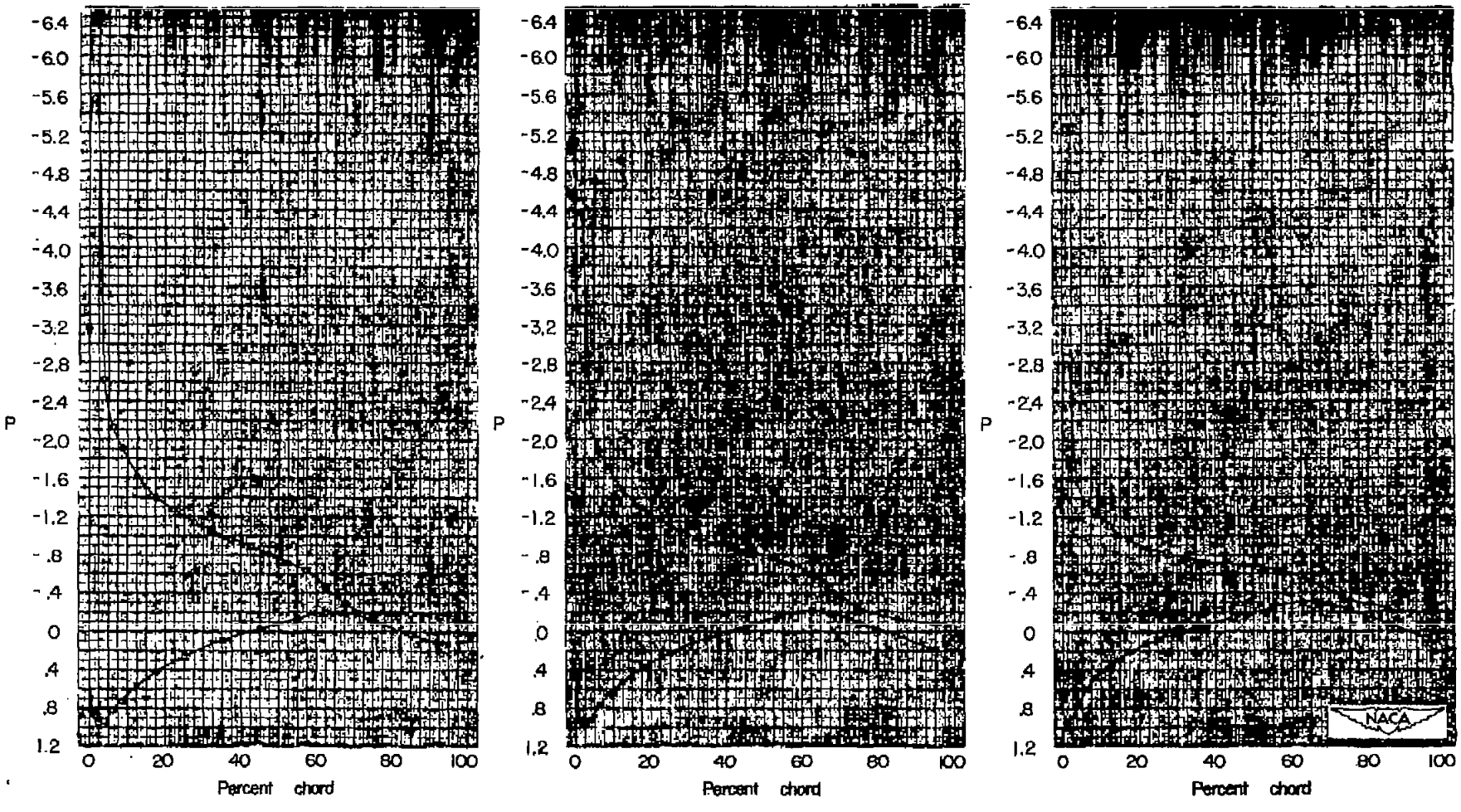
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(i) $\alpha = 15.22^{\circ}$.

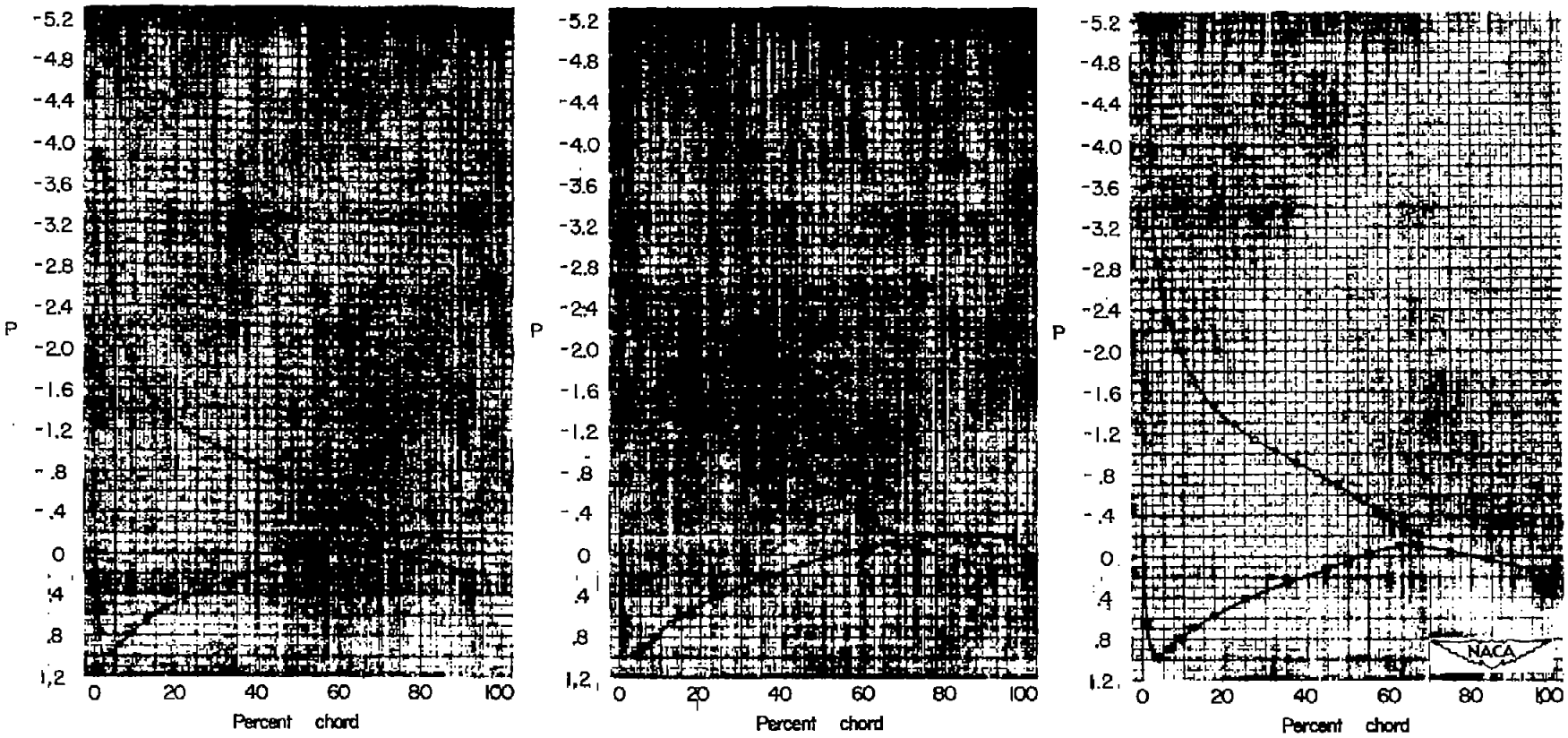
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(i) Concluded. $\alpha = 15.22^{\circ}$.

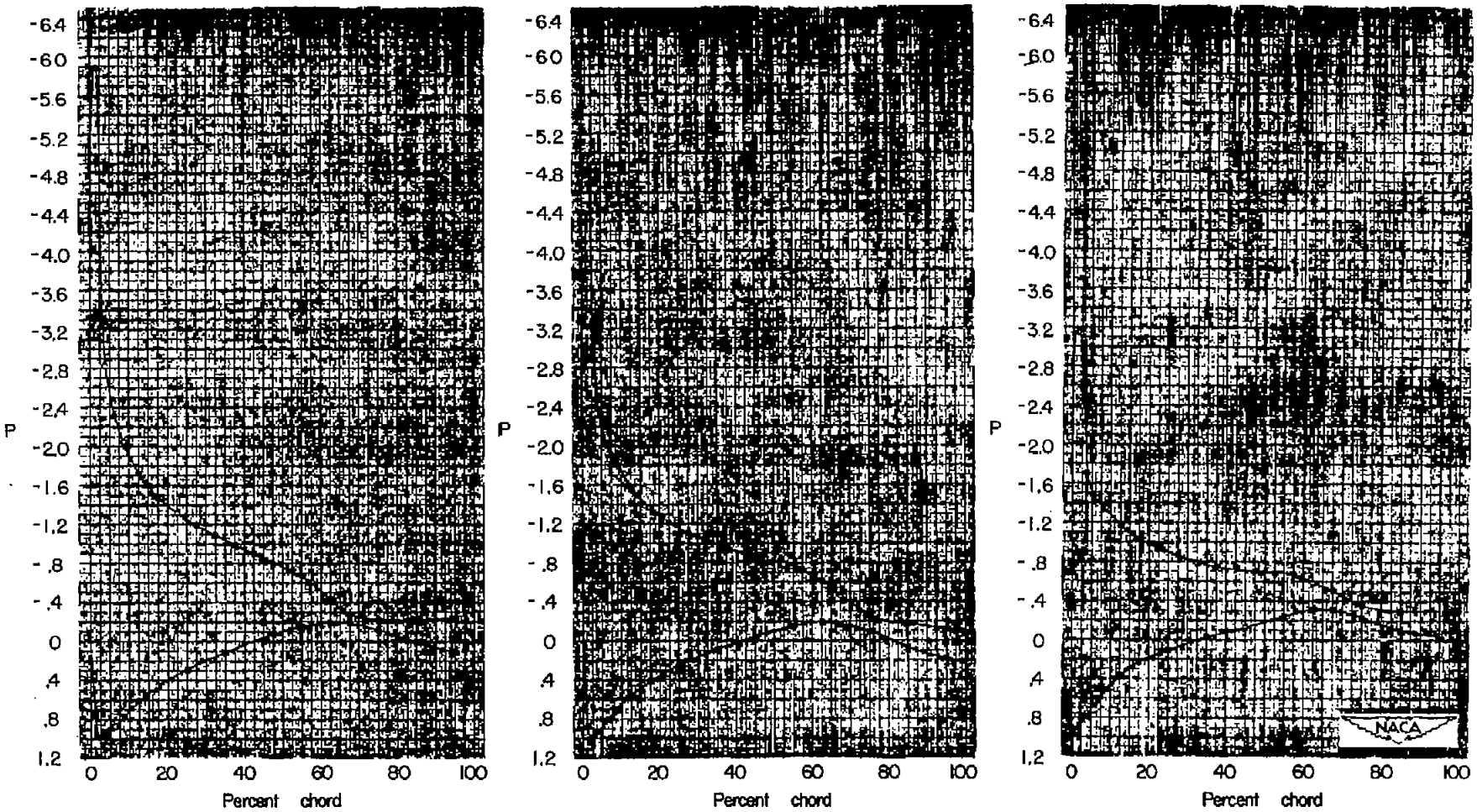
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(j) $\alpha = 16.27^{\circ}$.

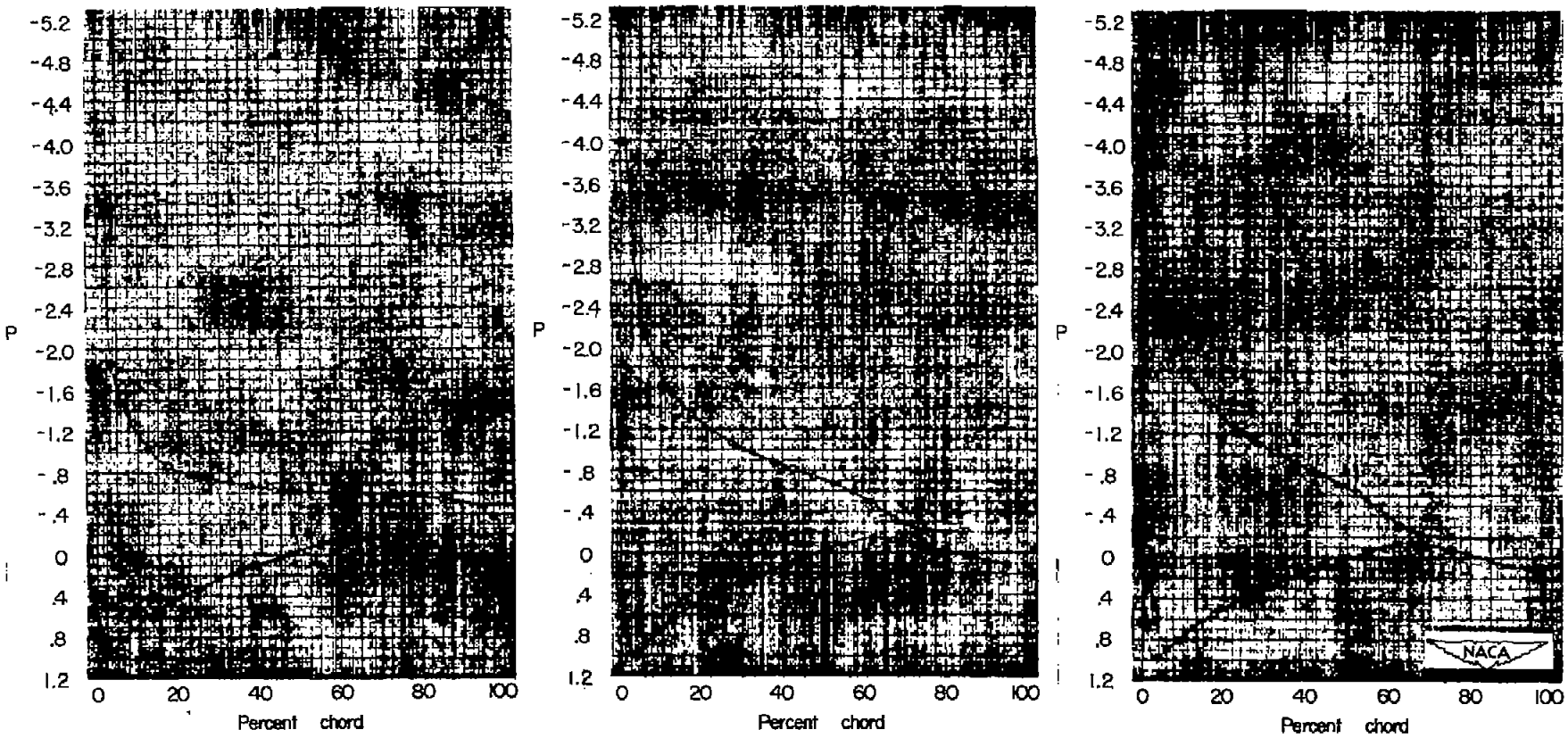
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(j) Concluded. $\alpha = 16.27^{\circ}$.

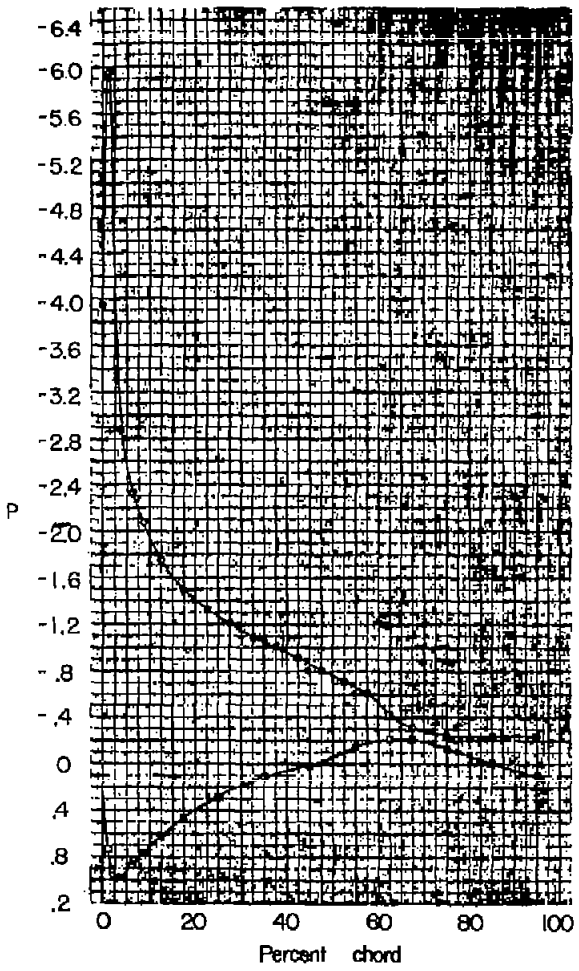
Figure 4.- Continued. $M = 0.20$.



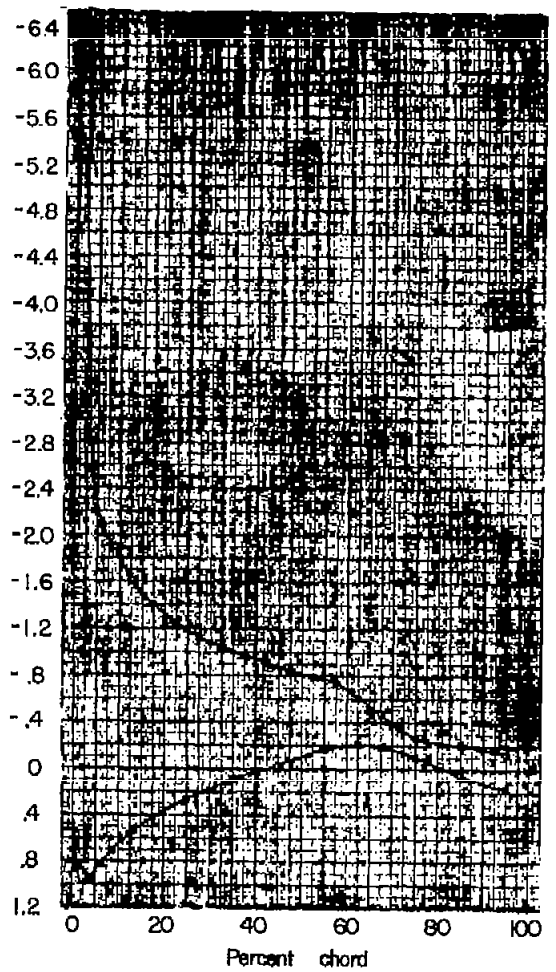
Station 1 Station 2 Station 3

(k) $\alpha = 17.10^{\circ}$.

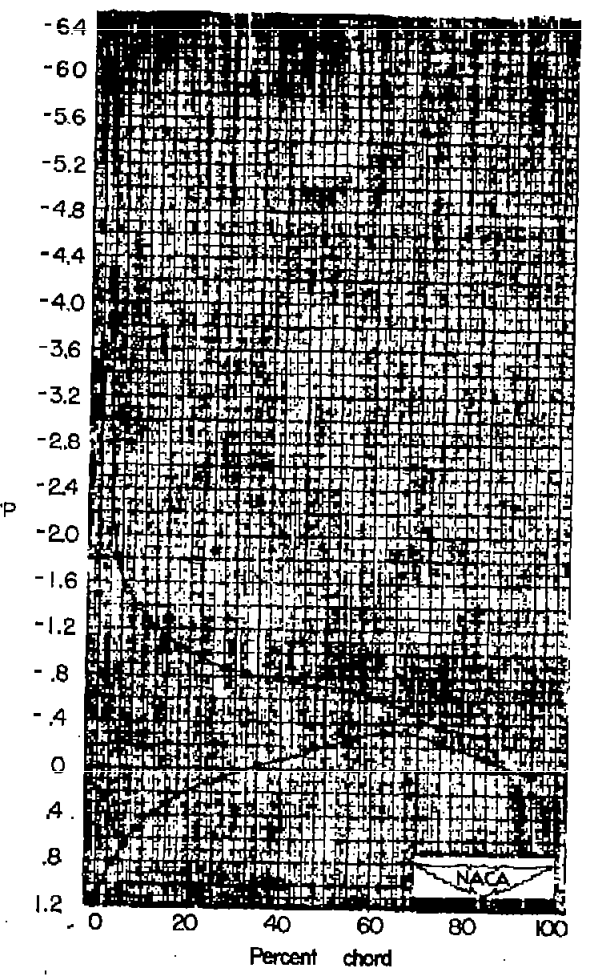
Figure 4.- Continued. M = 0.20.



Station 4



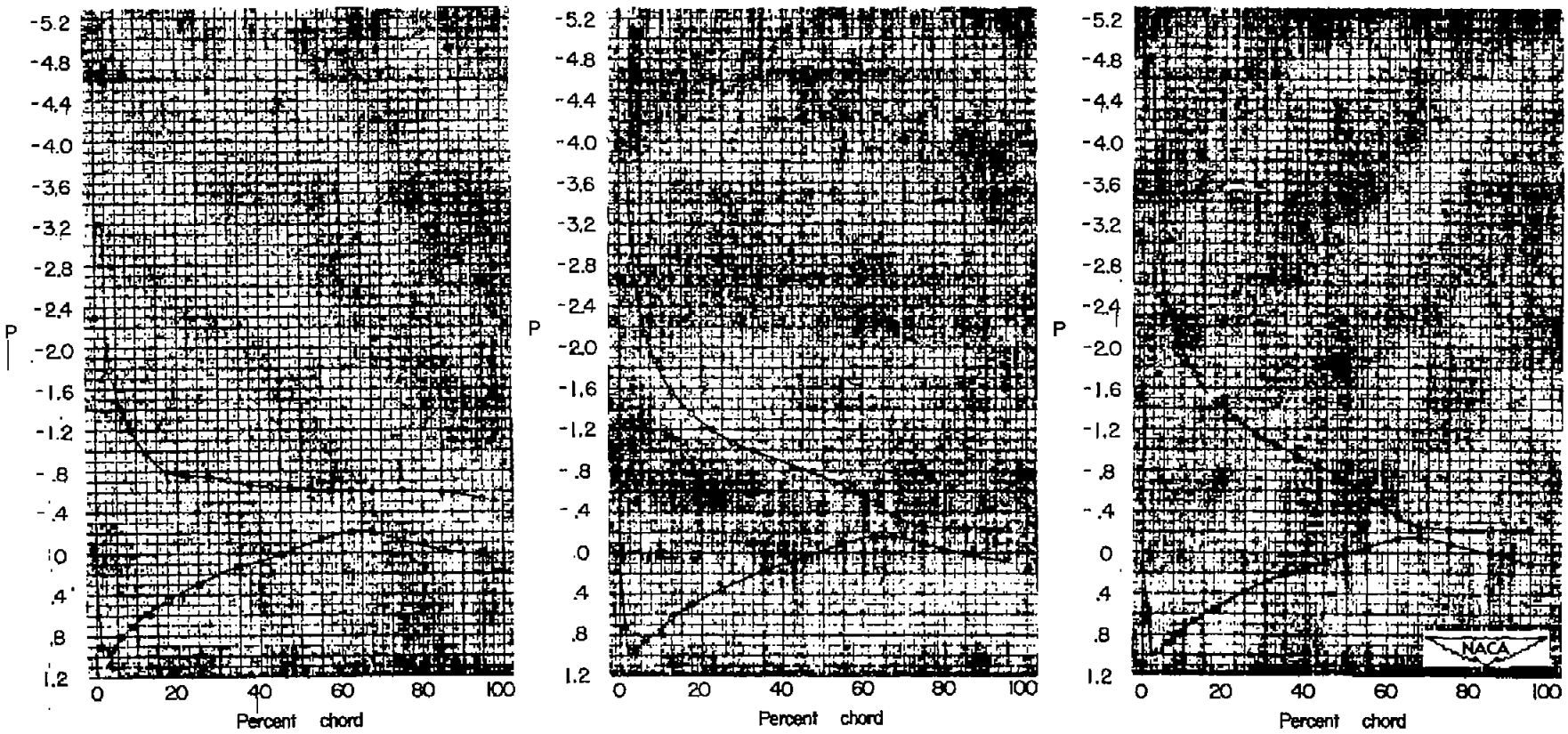
Station 5



Station 6

(k) Concluded. $\alpha = 17.10^{\circ}$.

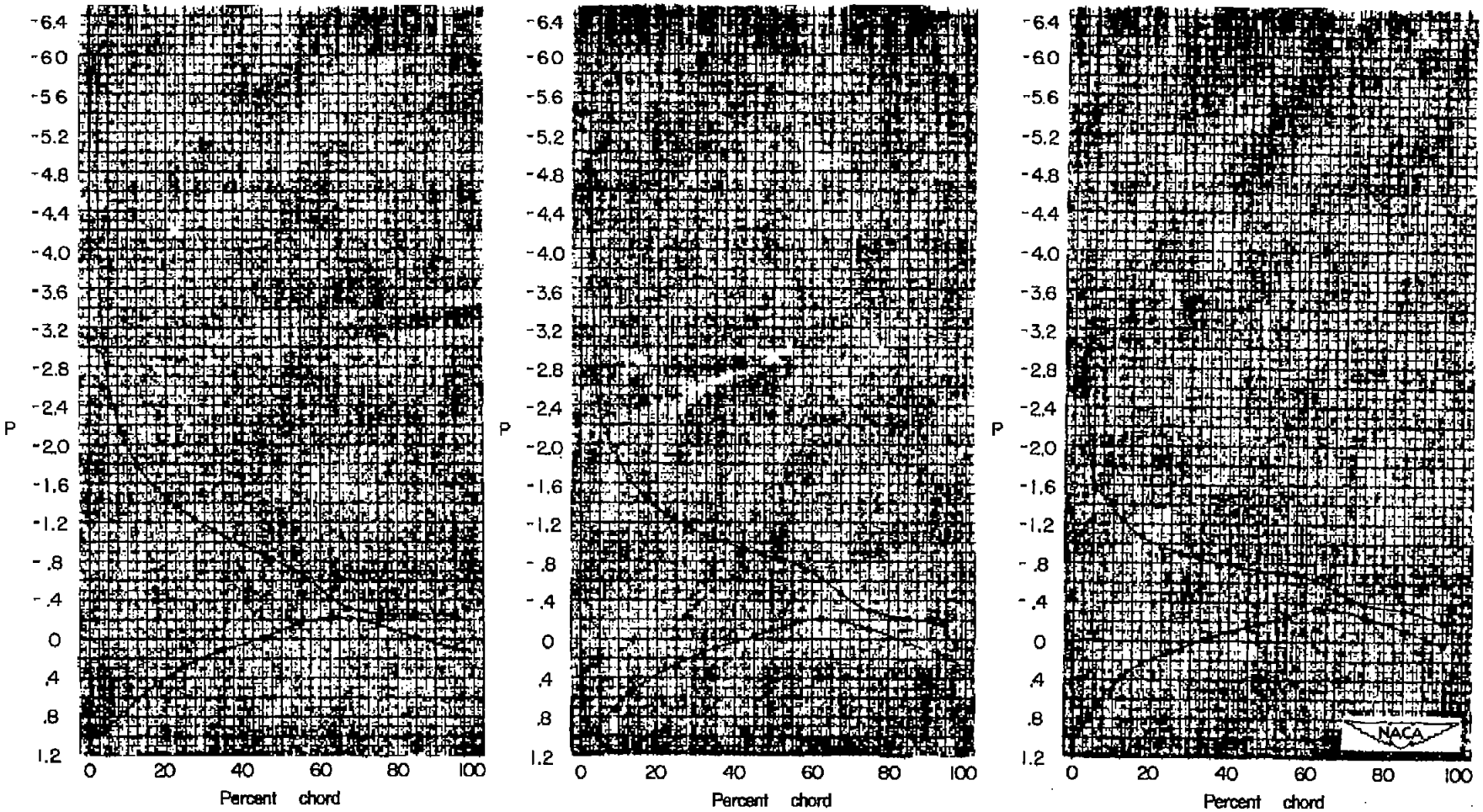
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(l) $\alpha = 17.52^{\circ}$.

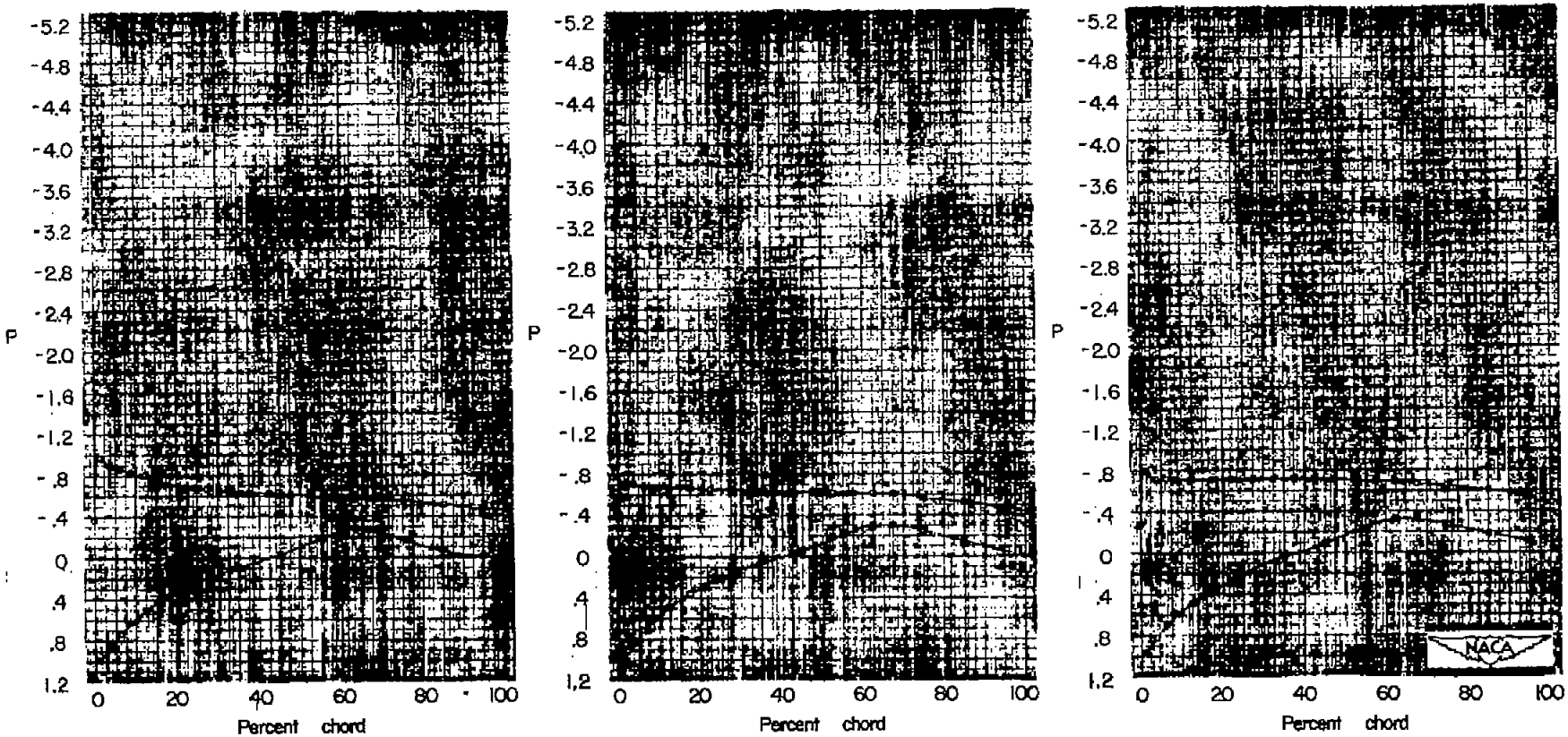
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(l) Concluded. $\alpha = 17.52^{\circ}$.

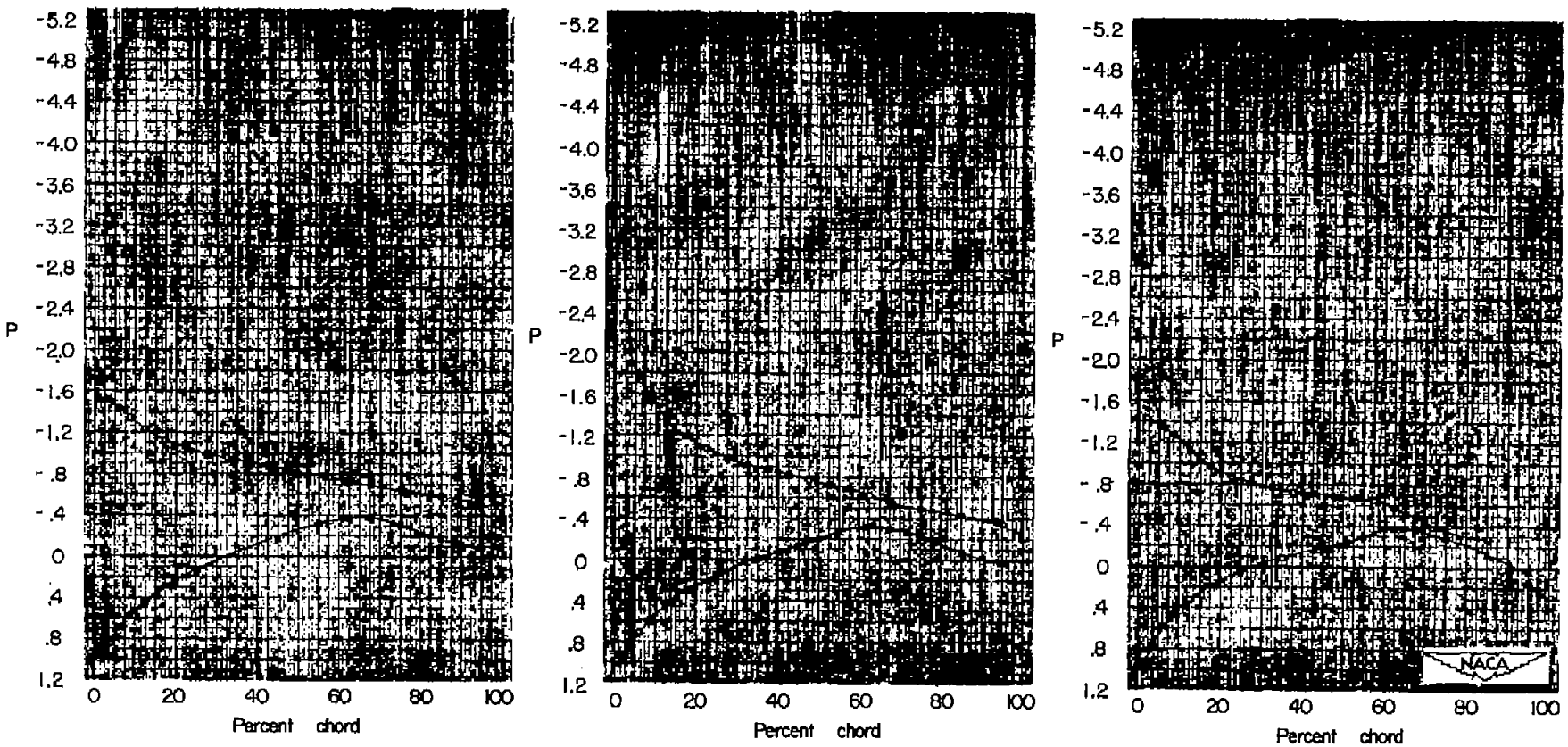
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(m) $\alpha = 17.52^{\circ}$.

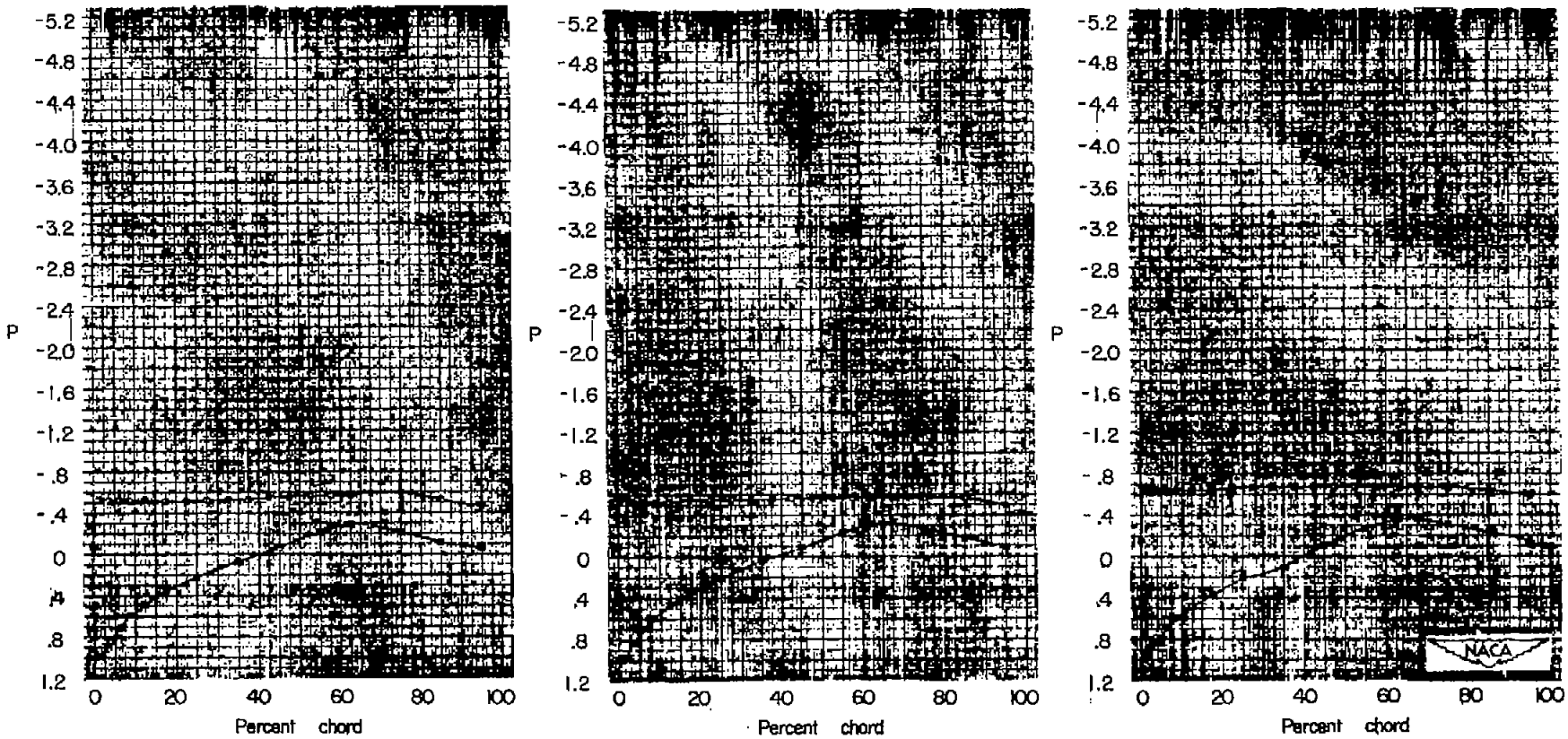
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(m) Concluded. $\alpha = 17.52^{\circ}$.

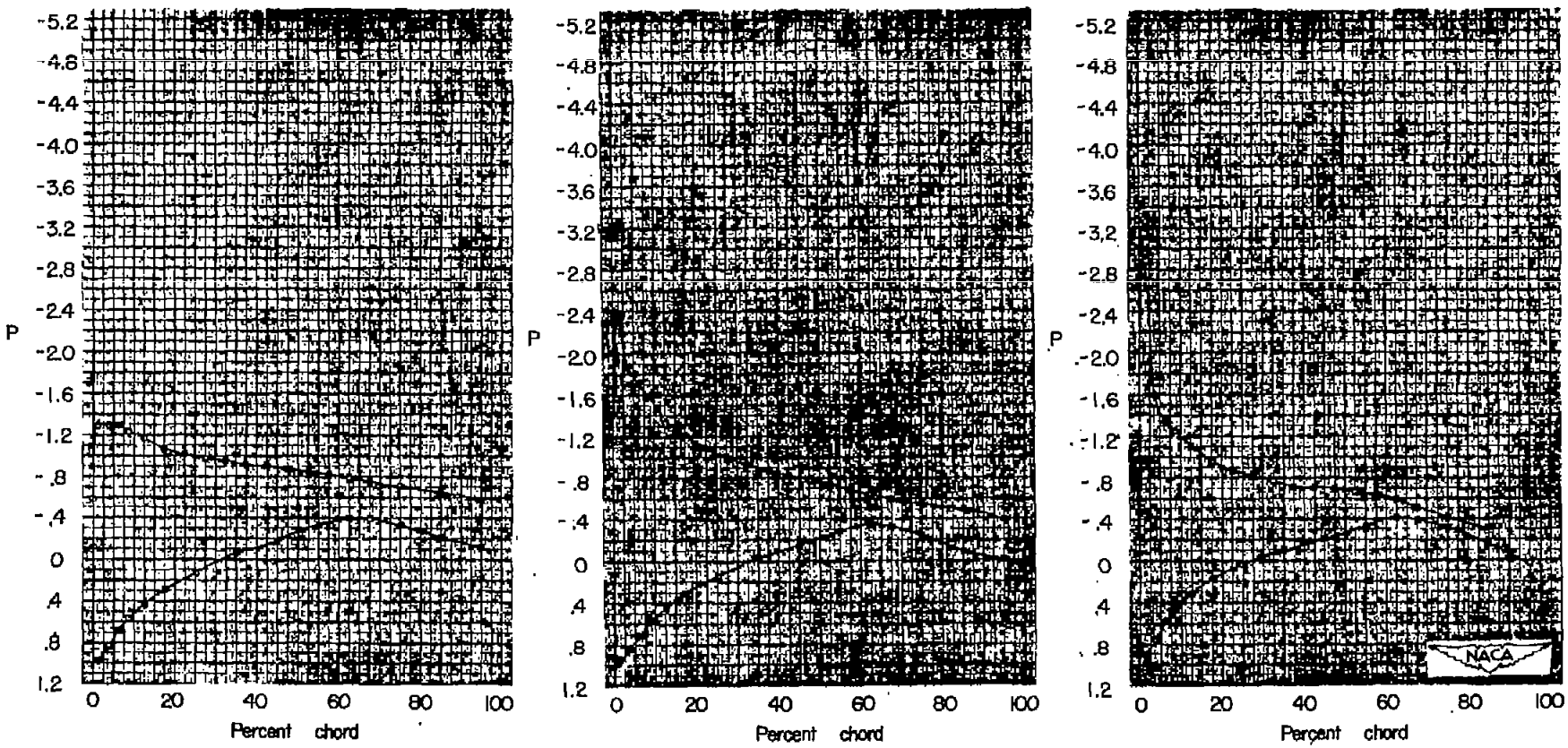
Figure 4.- Continued. M = 0.20.



Station 1 Station 2 Station 3

(n) $\alpha = 18.35^{\circ}$.

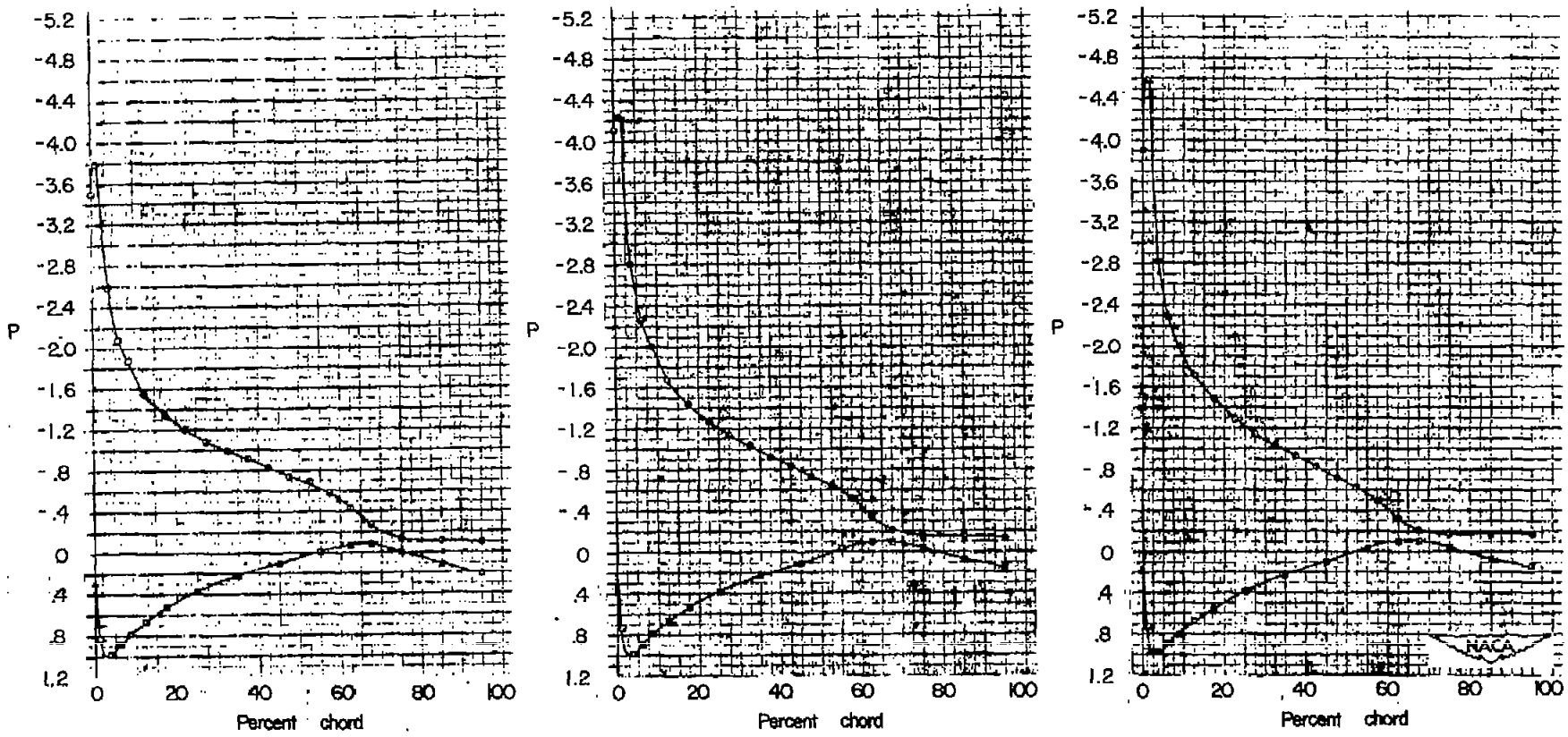
Figure 4.- Continued. M = 0.20.



Station 4 Station 5 Station 6

(n) Concluded. $\alpha = 18.35^{\circ}$.

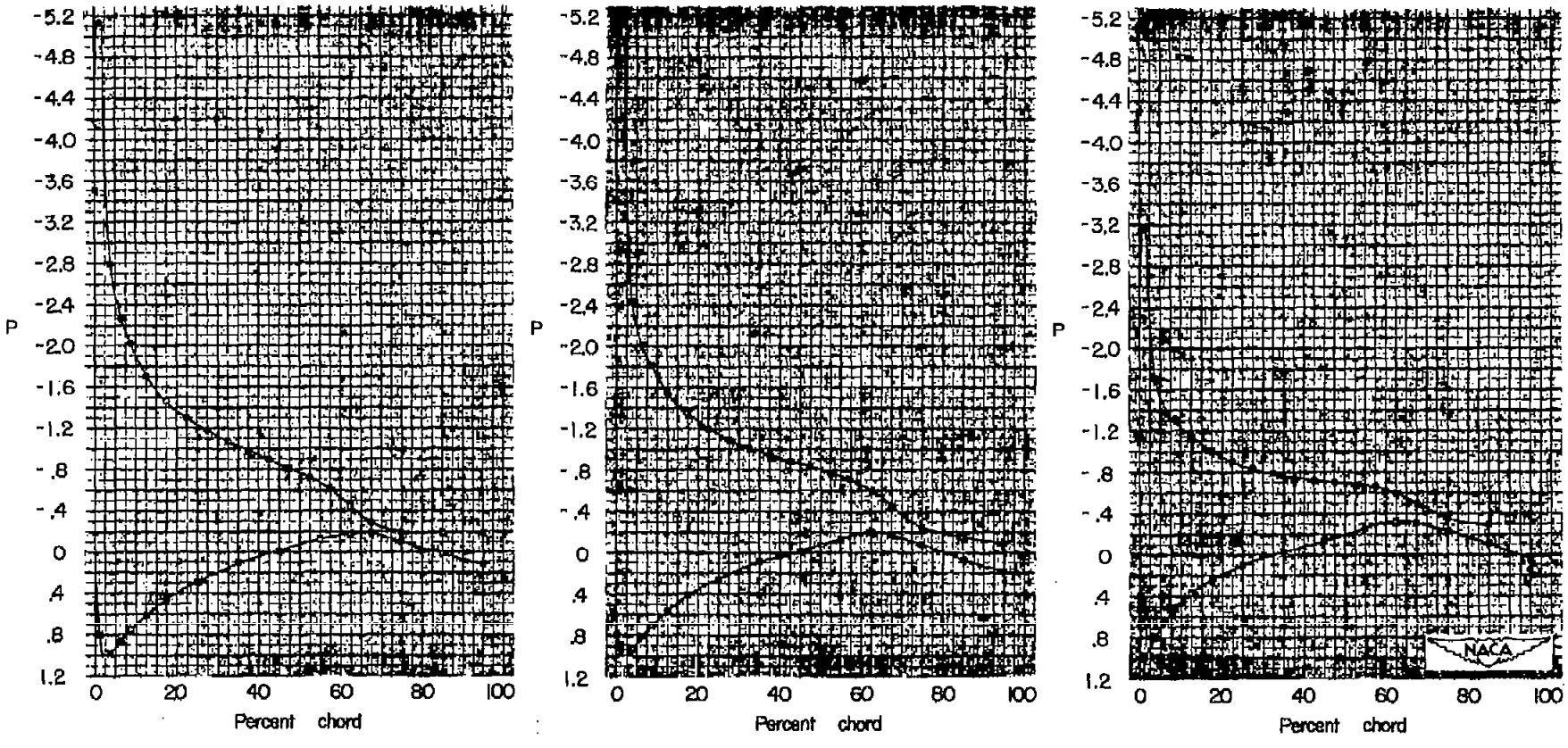
Figure 4.- Concluded. M = 0.20.



Station 1 Station 2 Station 3

(a) $\alpha = 15.84^{\circ}$.

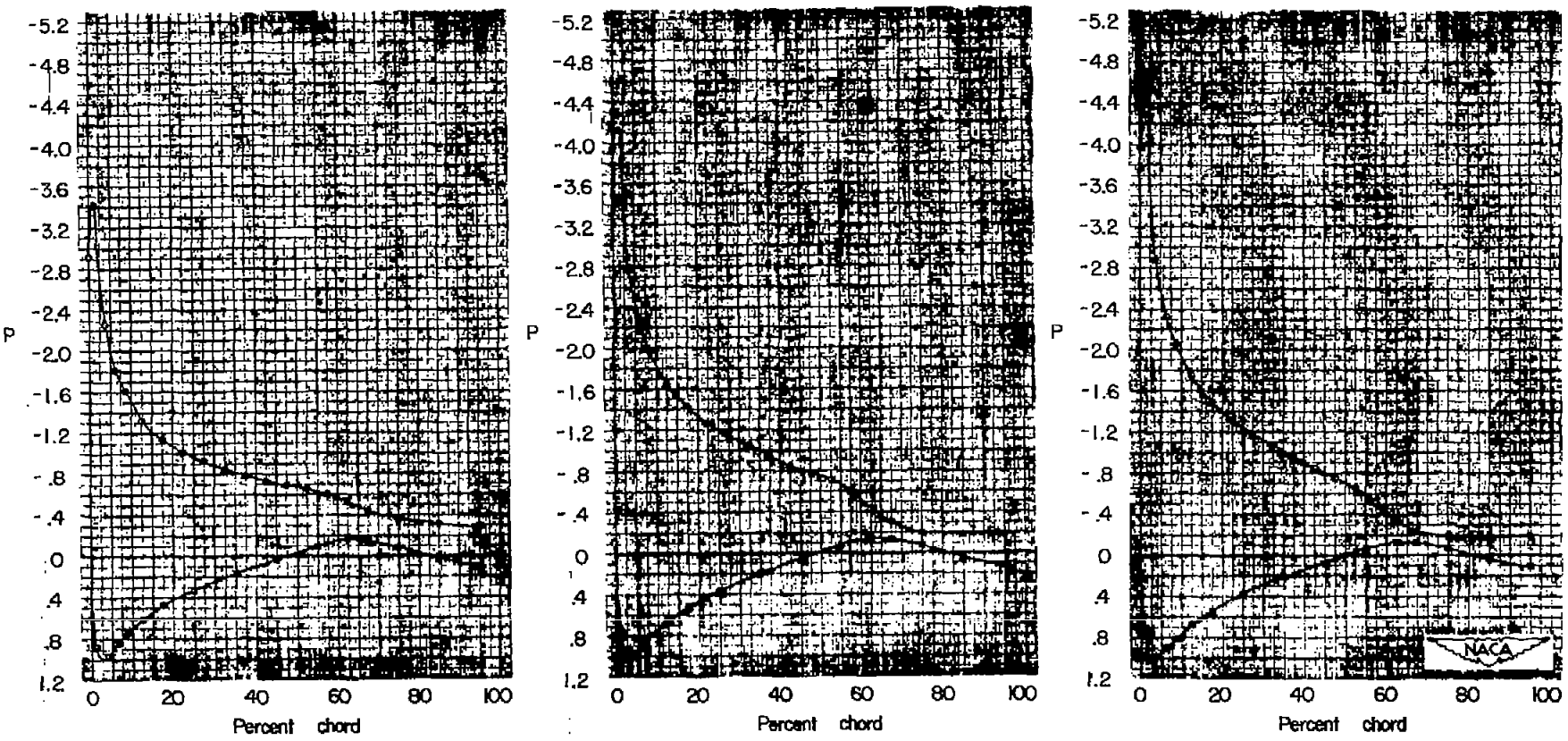
Figure 5.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.25.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = 15.84^{\circ}$.

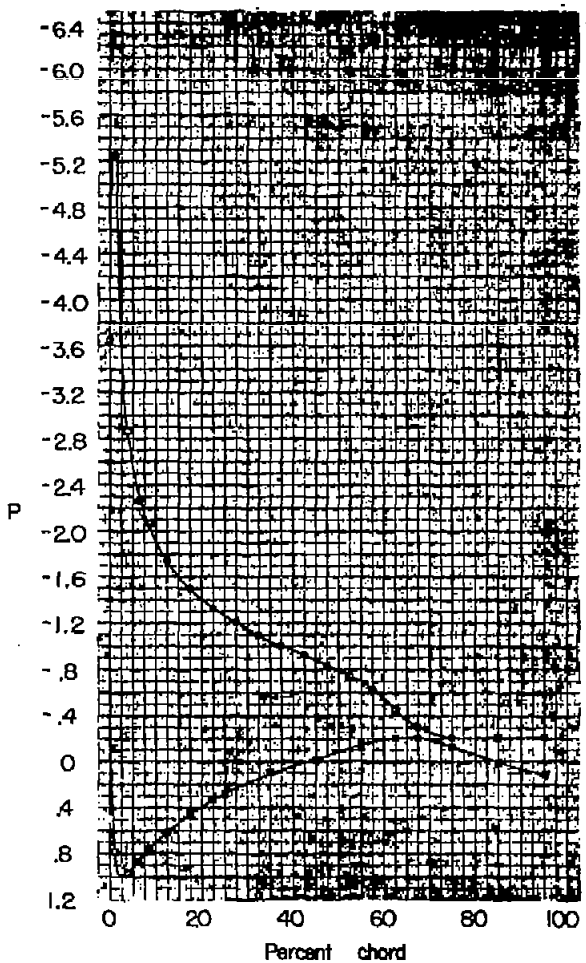
Figure 5.- Continued. M = 0.25.



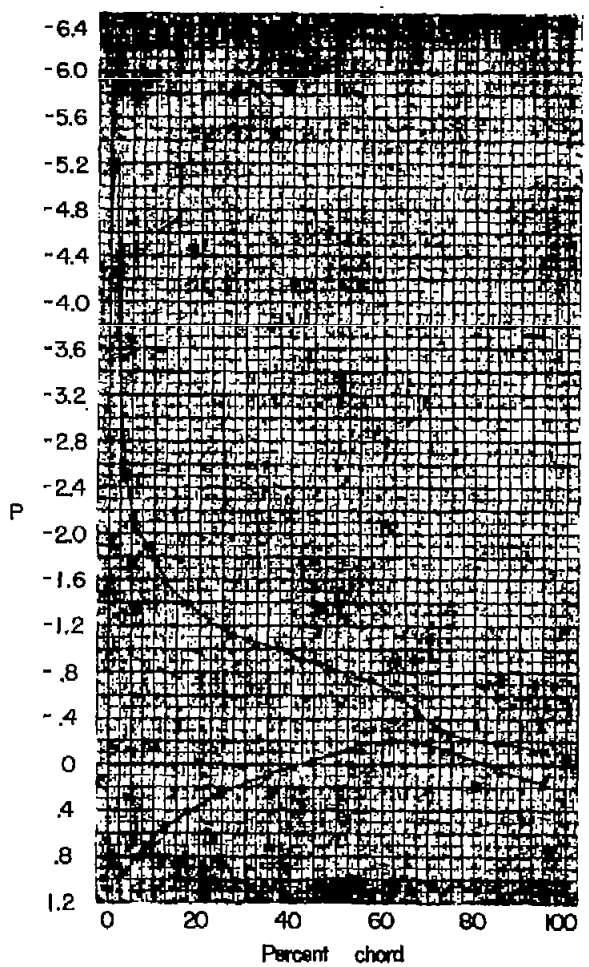
Station 1 Station 2 Station 3

(b) $\alpha = 16.68^{\circ}$.

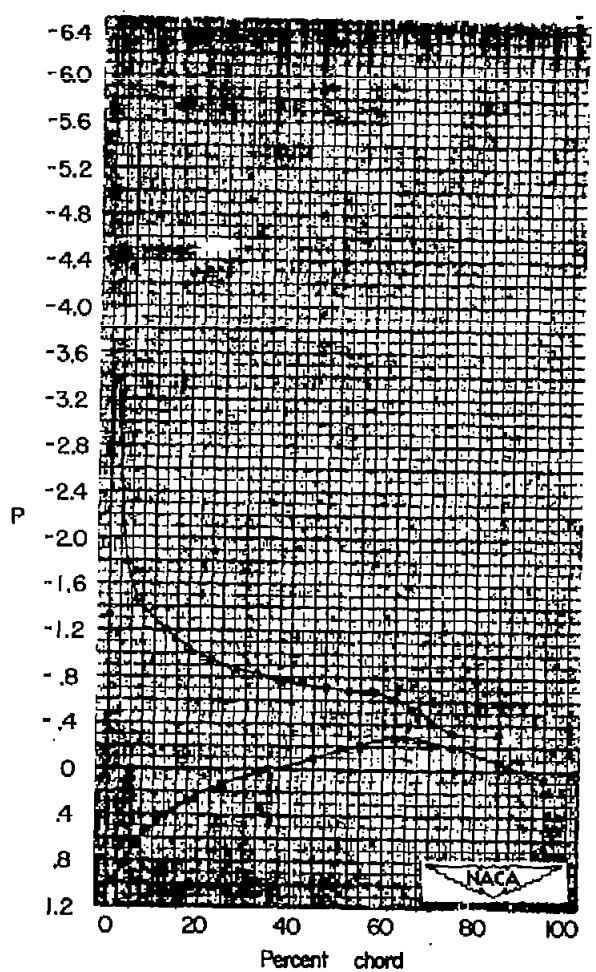
Figure 5.- Continued. M = 0.25.



Station 4



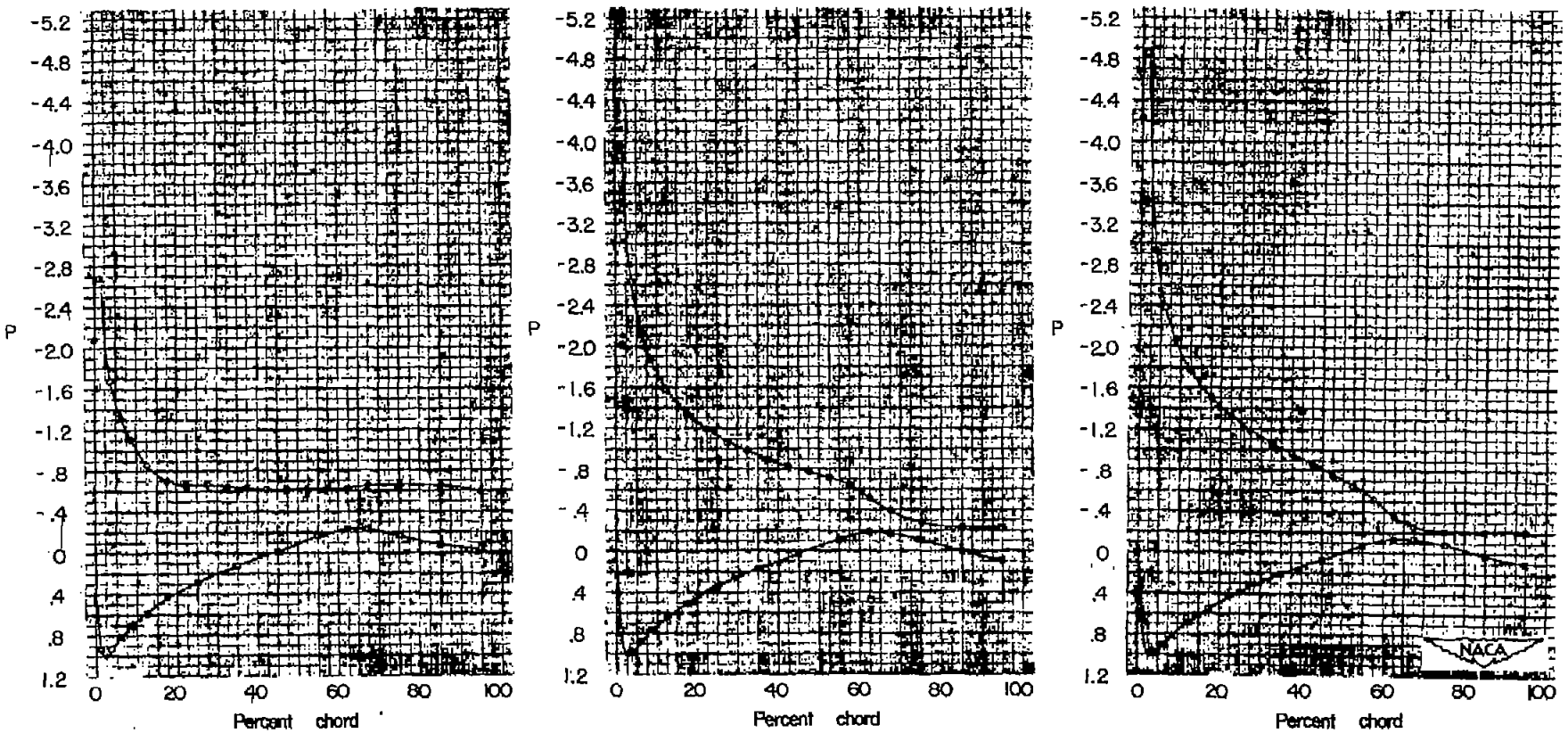
Station 5



Station 6

(b) Concluded. $\alpha = 16.68^{\circ}$.

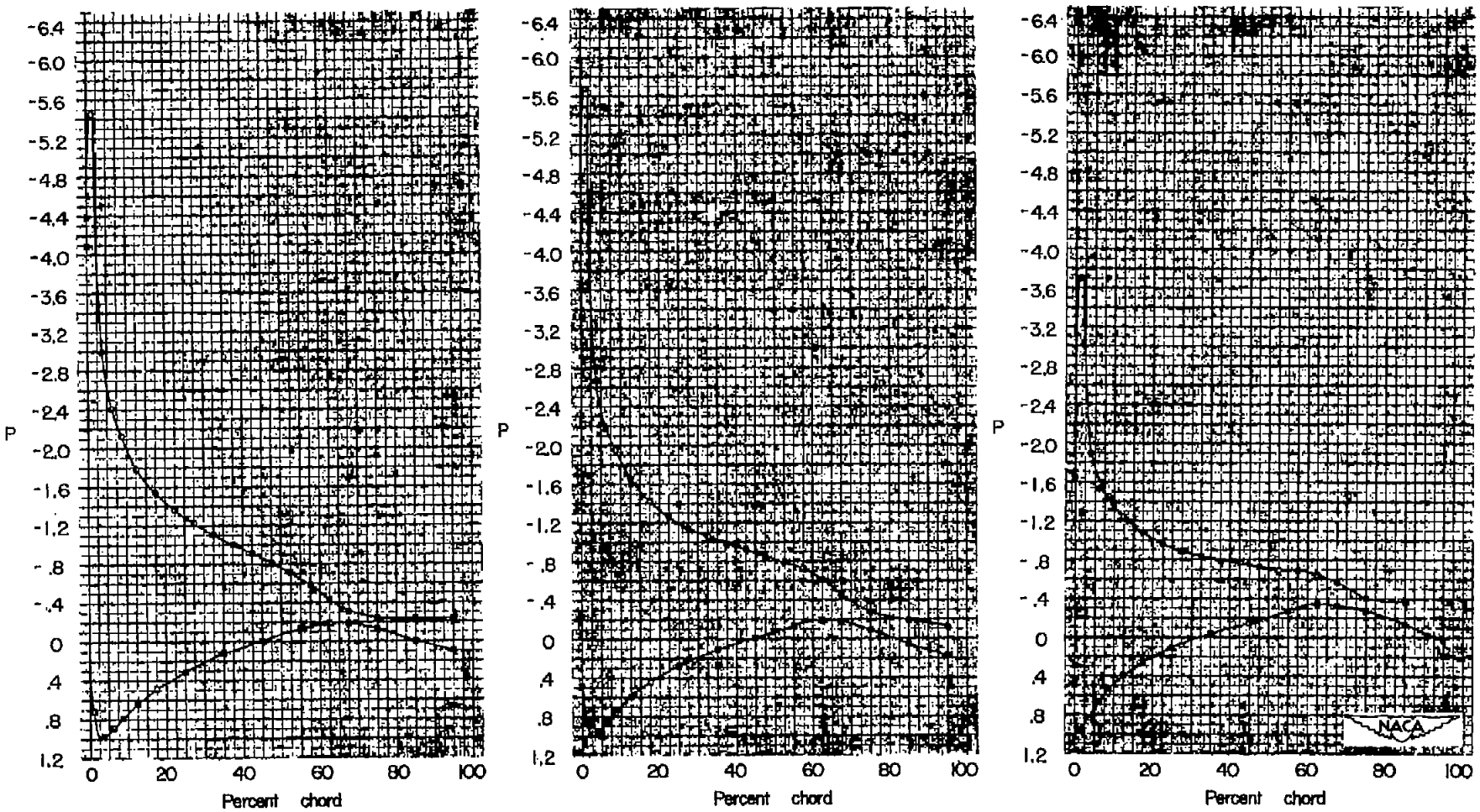
Figure 5.- Continued. M = 0.25.



Station 1 Station 2 Station 3

(c) $\alpha = 17.83^{\circ}$.

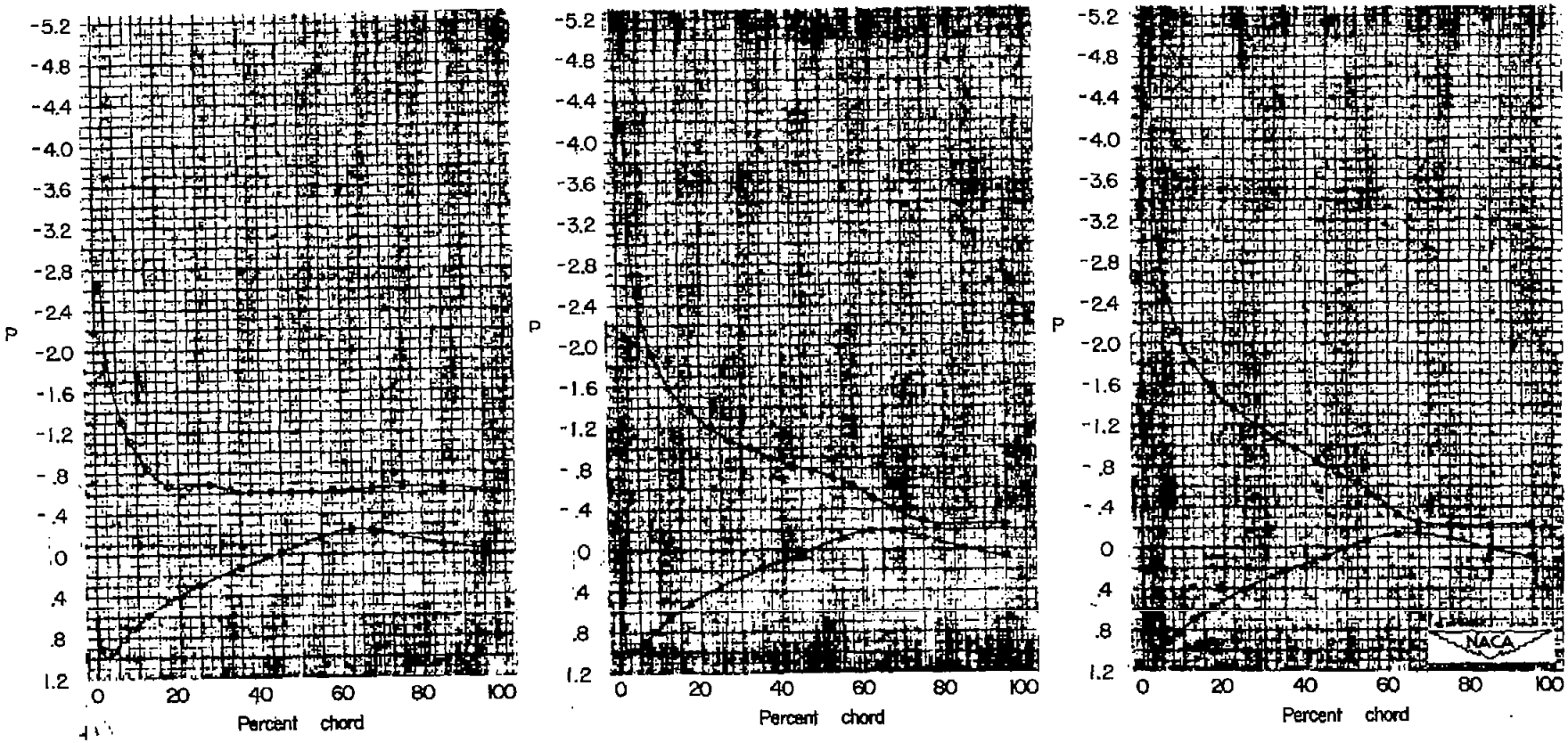
Figure 5.- Continued. M = 0.25.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 17.83^{\circ}$.

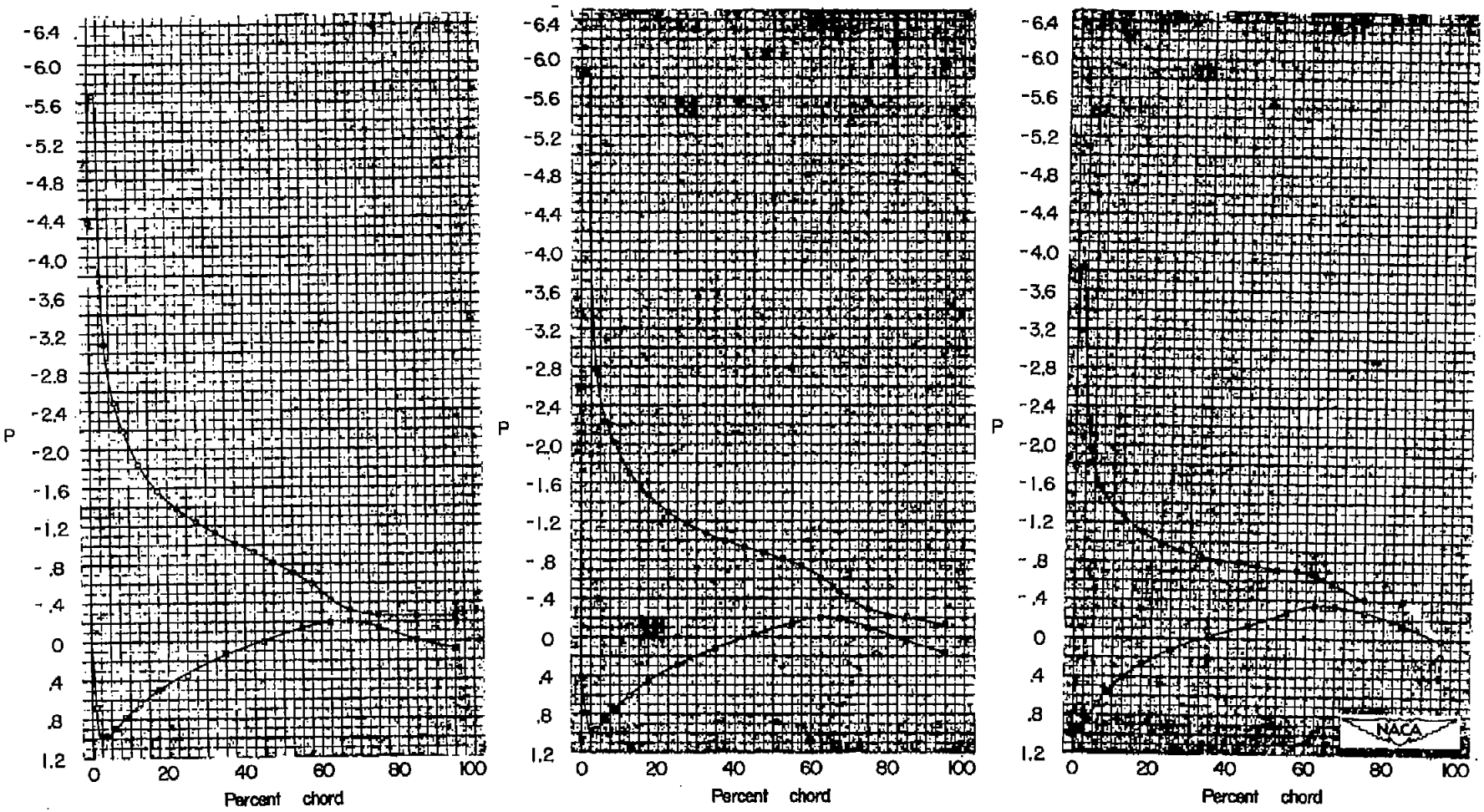
Figure 5.- Continued. M = 0.25.



Station 1 Station 2 Station 3

(d) $\alpha = 18.35^{\circ}$.

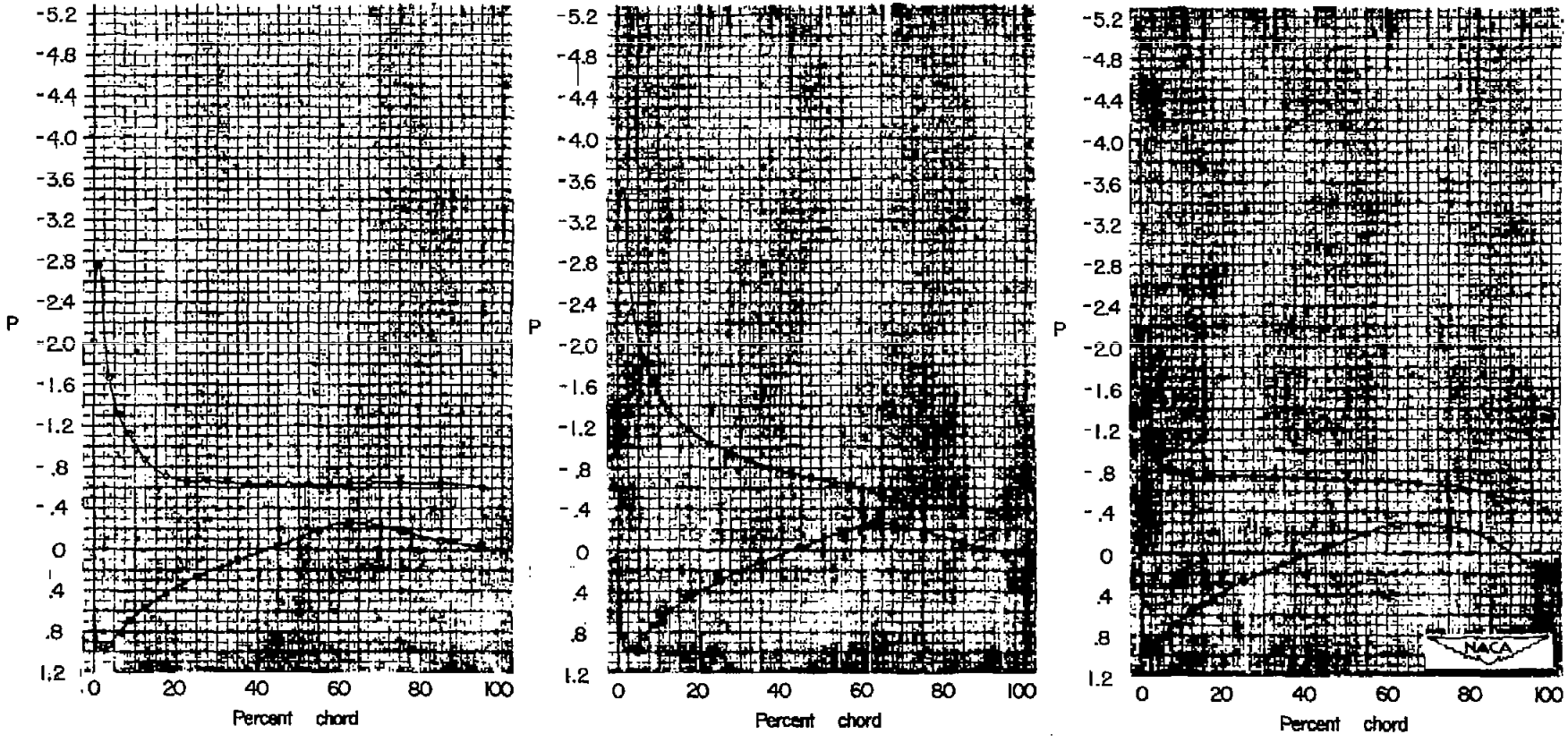
Figure 5.- Continued. M = 0.25.



Station 4 Station 5 Station 6

(d) Concluded. $\alpha = 18.35^{\circ}$.

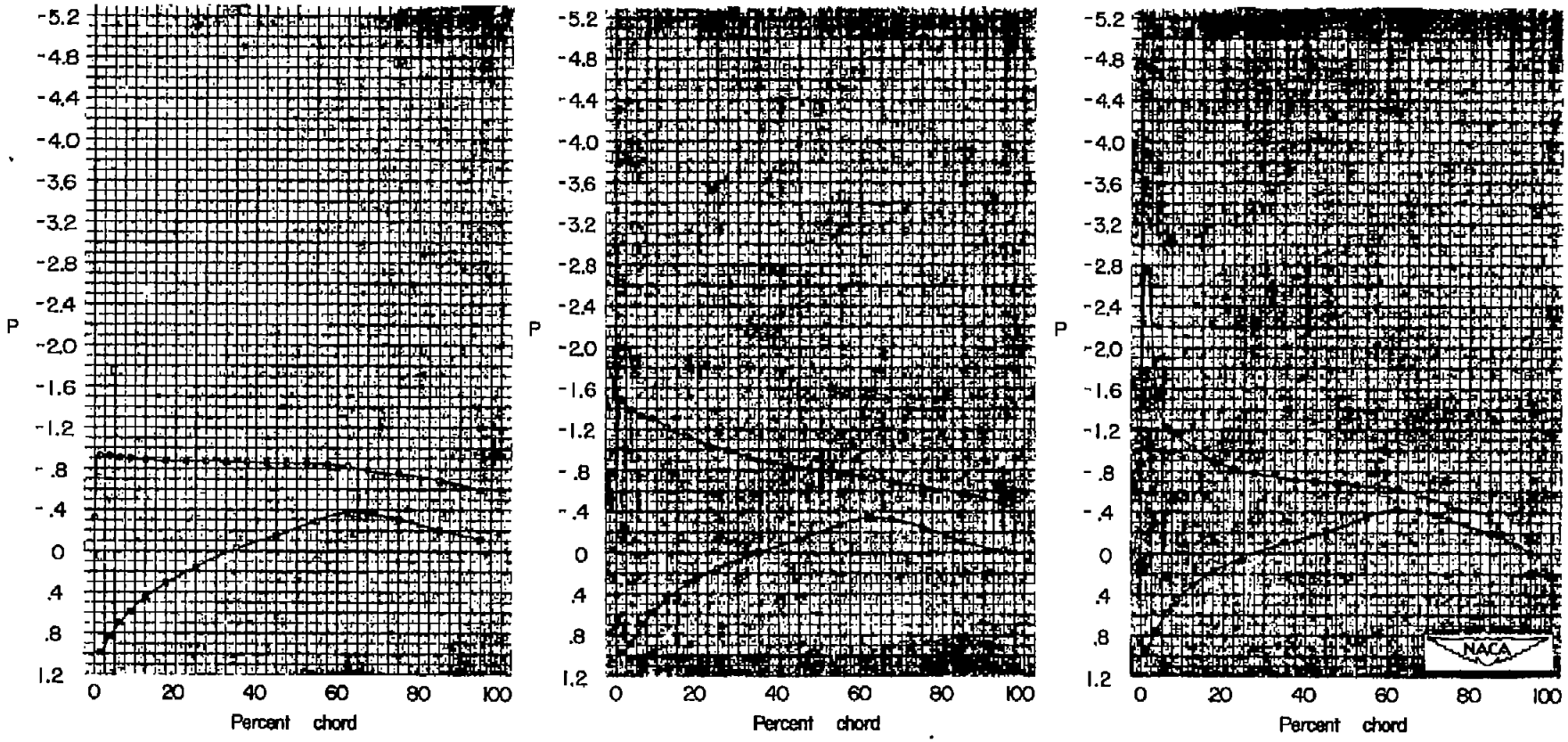
Figure 5.- Continued. M = 0.25.



Station 1 Station 2 Station 3

(e) $\alpha = 18.35^{\circ}$.

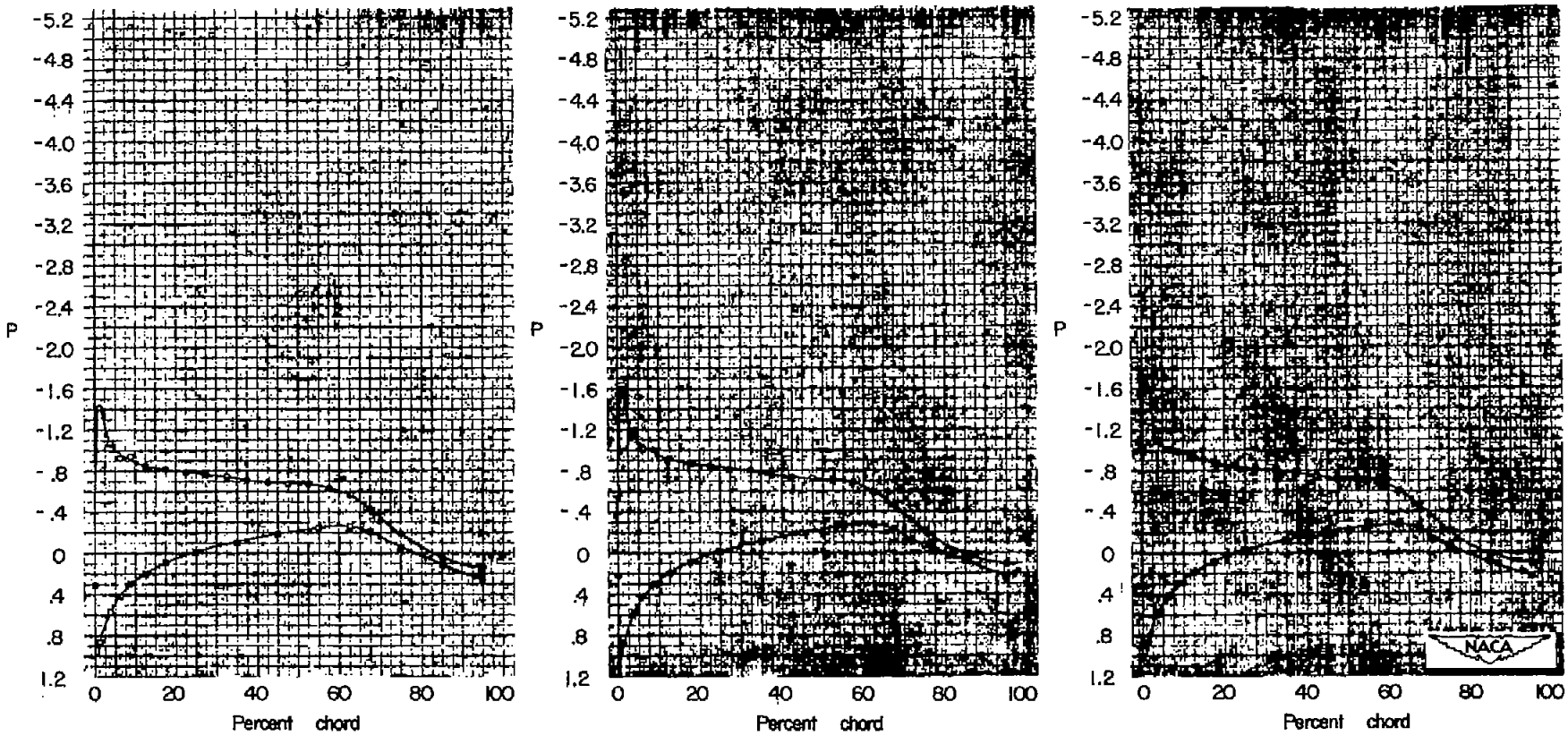
Figure 5.- Continued. M = 0.25.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 18.35^{\circ}$.

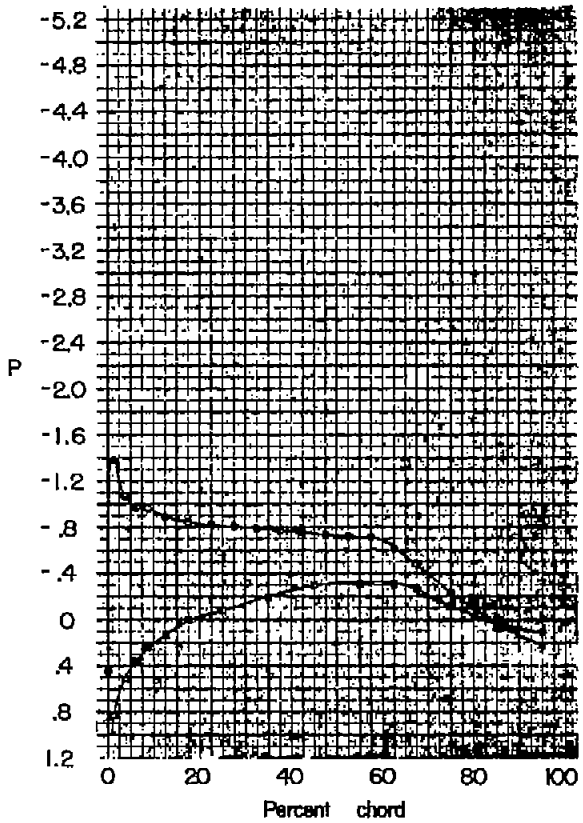
Figure 5.- Concluded. M = 0.25.



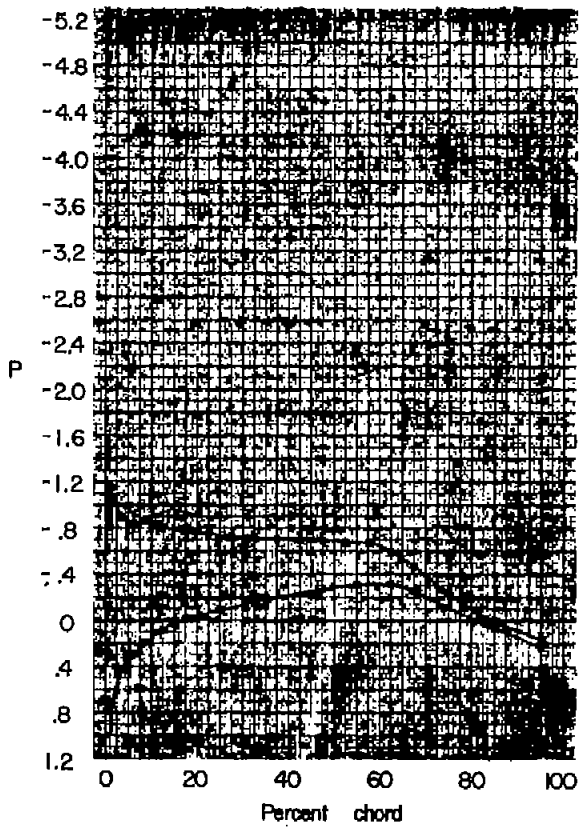
Station 1 Station 2 Station 3

(a) $\alpha = 6.72^{\circ}$.

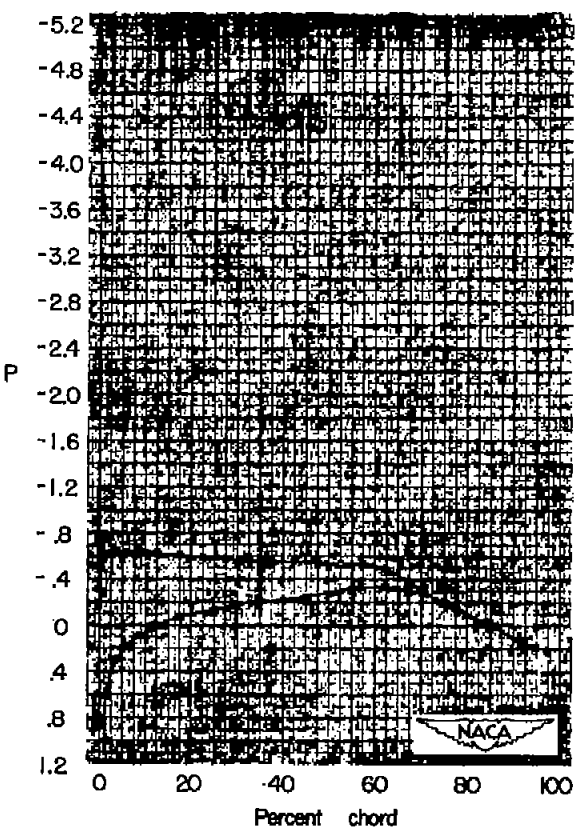
Figure 6.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.30.



Station 4



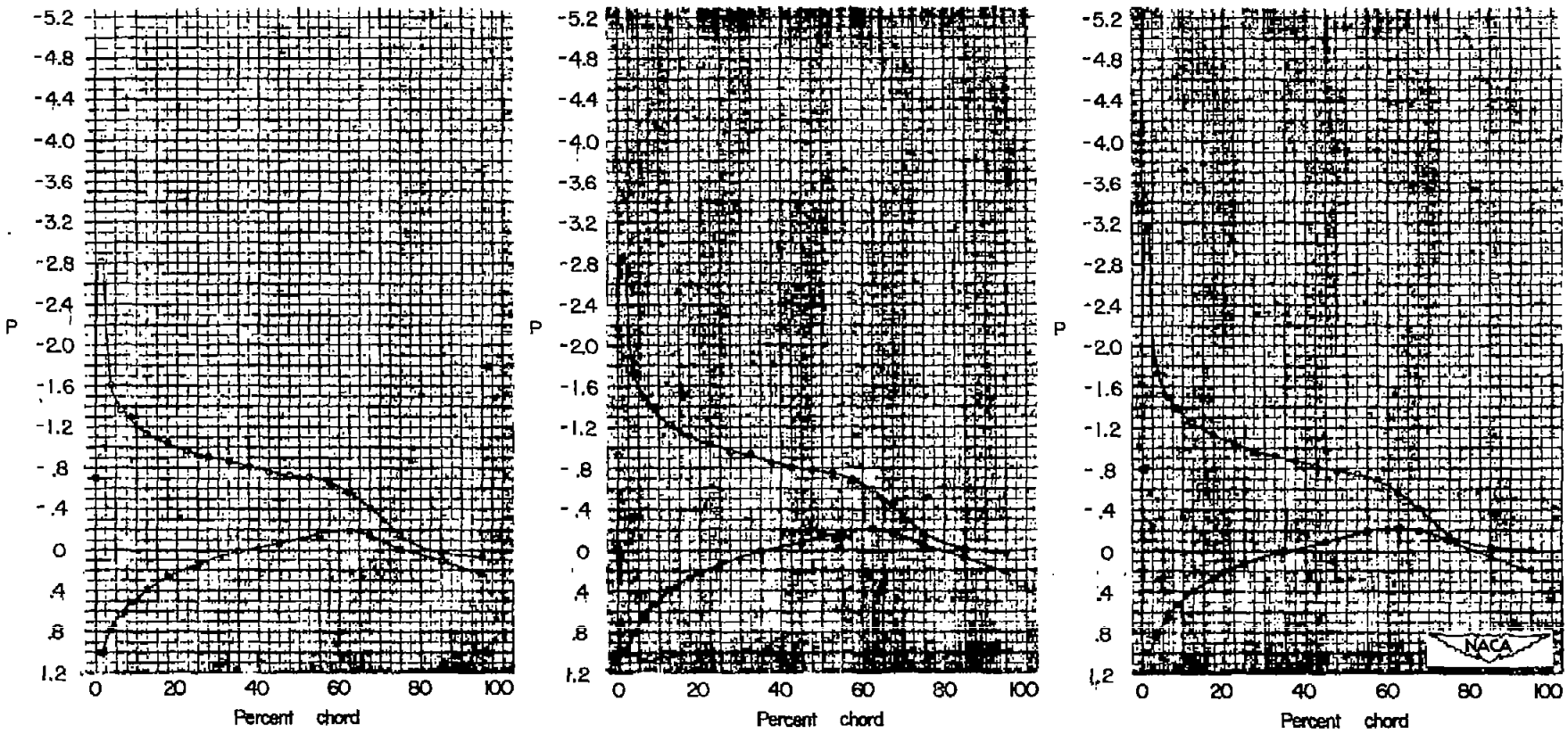
Station 5



Station 6

(a) Concluded. $\alpha = 6.72^{\circ}$.

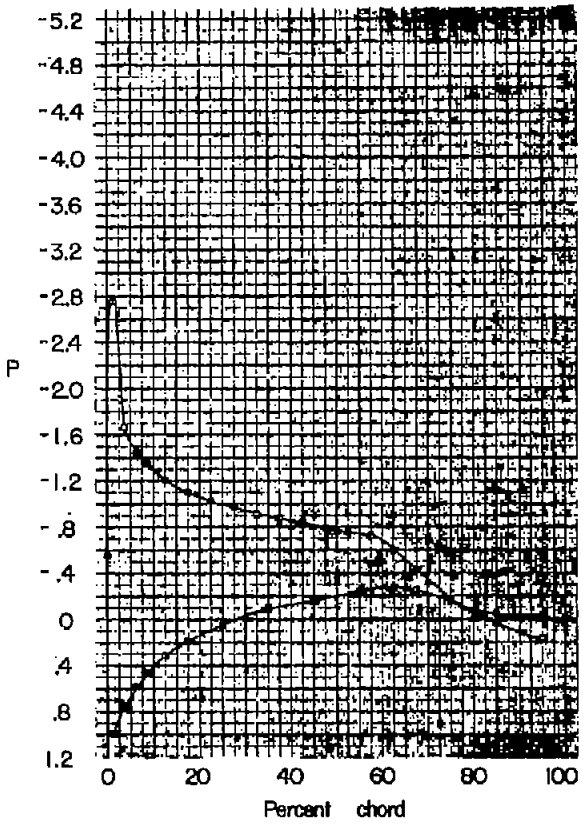
Figure 6.- Continued. M = 0.30.



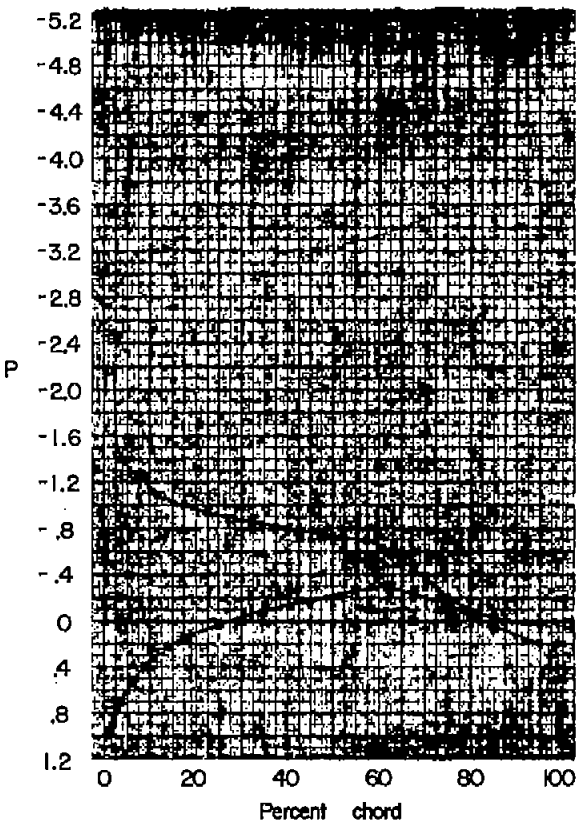
Station 1 Station 2 Station 3

(b) $\alpha = 9.92^{\circ}$.

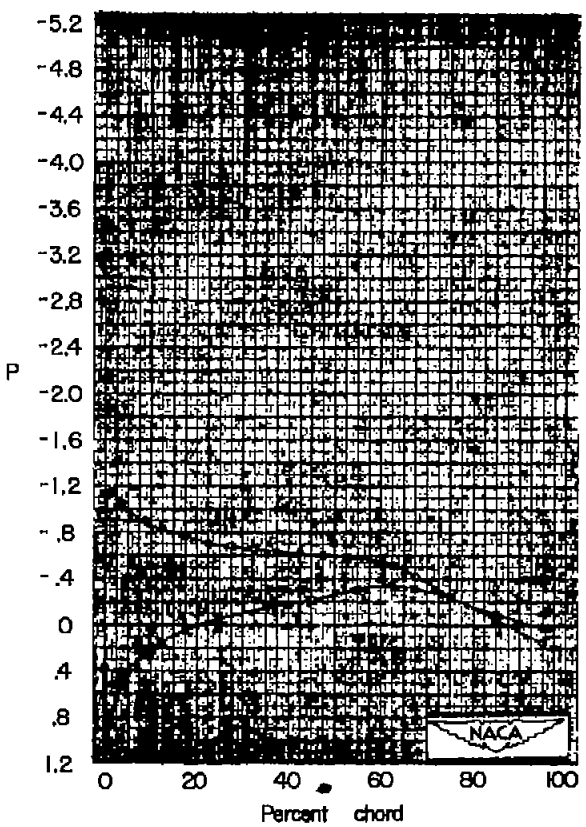
Figure 6.- Continued. M = 0.30.



Station 4



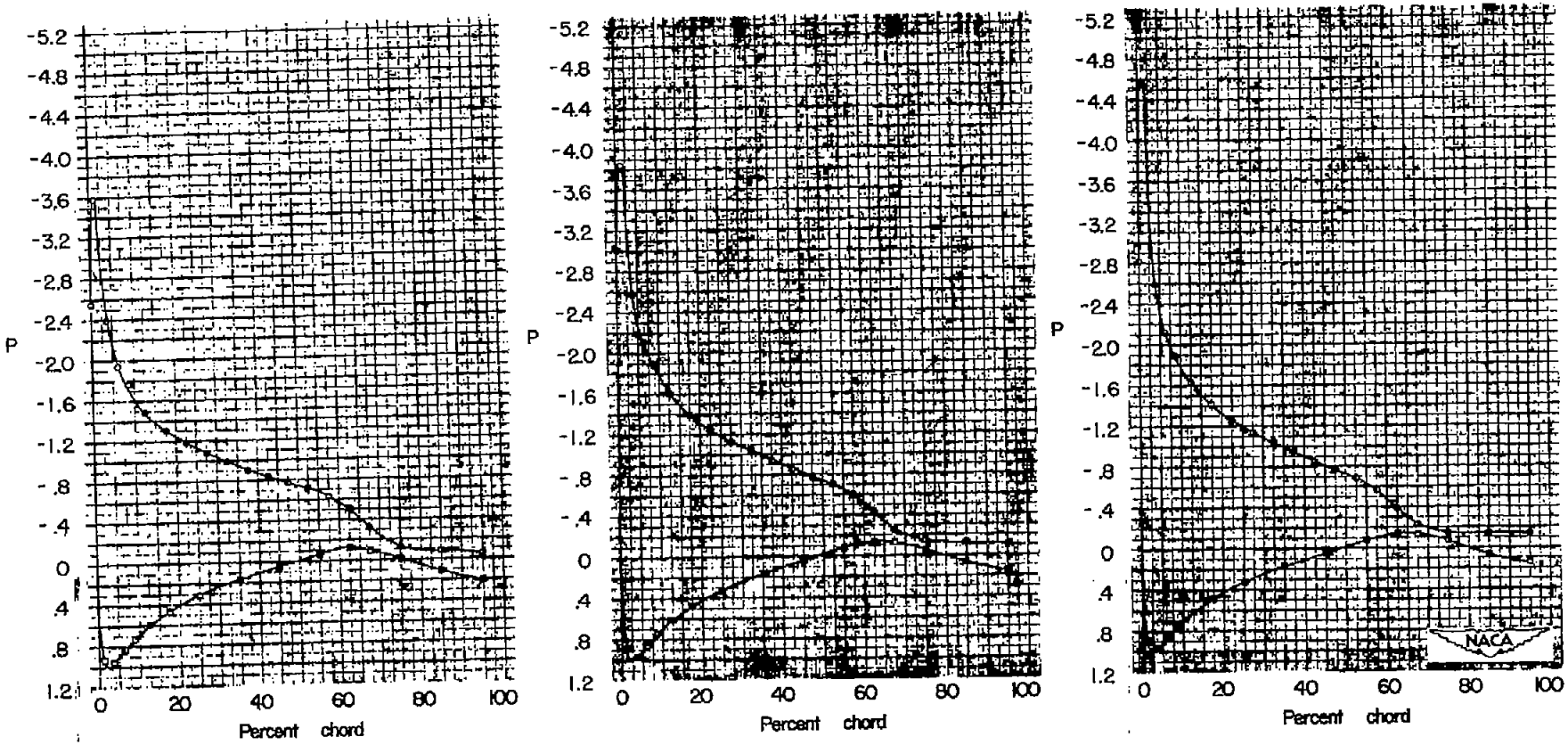
Station 5



Station 6

(b) Concluded. $\alpha = 9.92^{\circ}$.

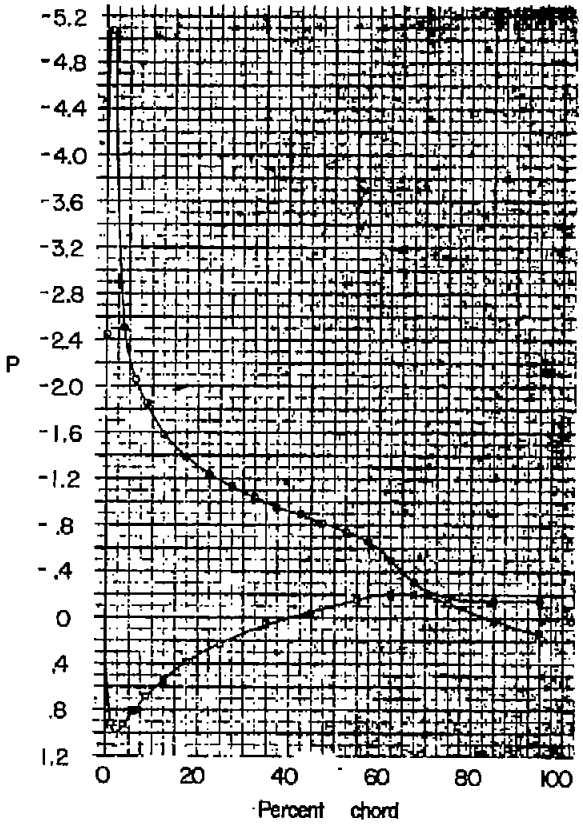
Figure 6.- Continued. M = 0.30.



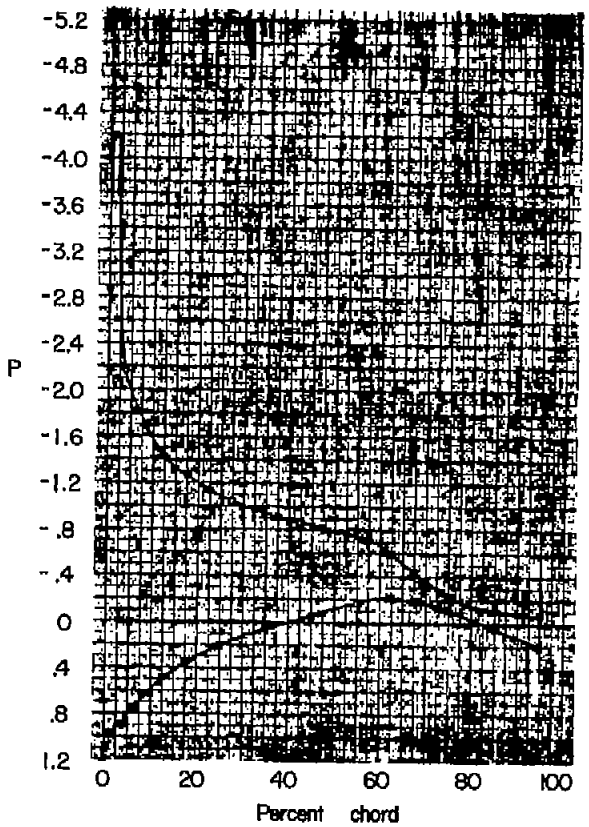
Station 1 Station 2 Station 3

(c) $\alpha = 14.17^{\circ}$.

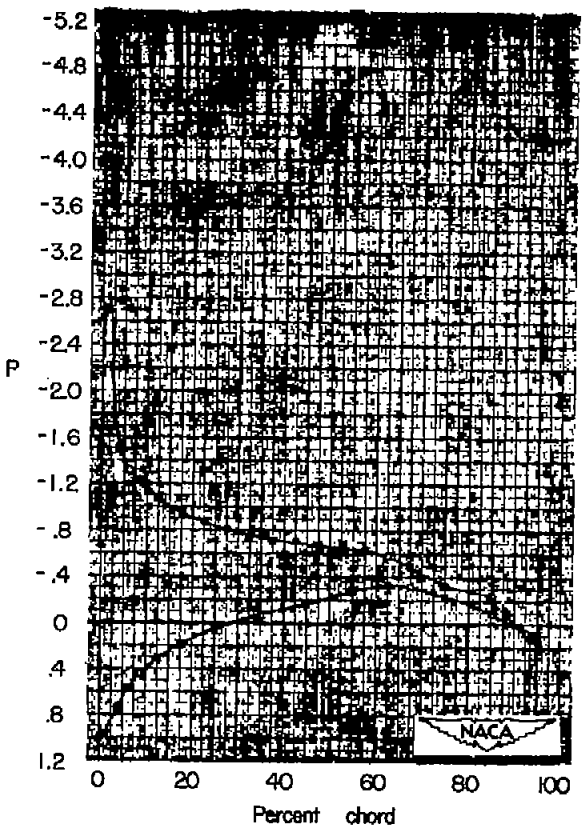
Figure 6.- Continued. M = 0.30.



Station 4



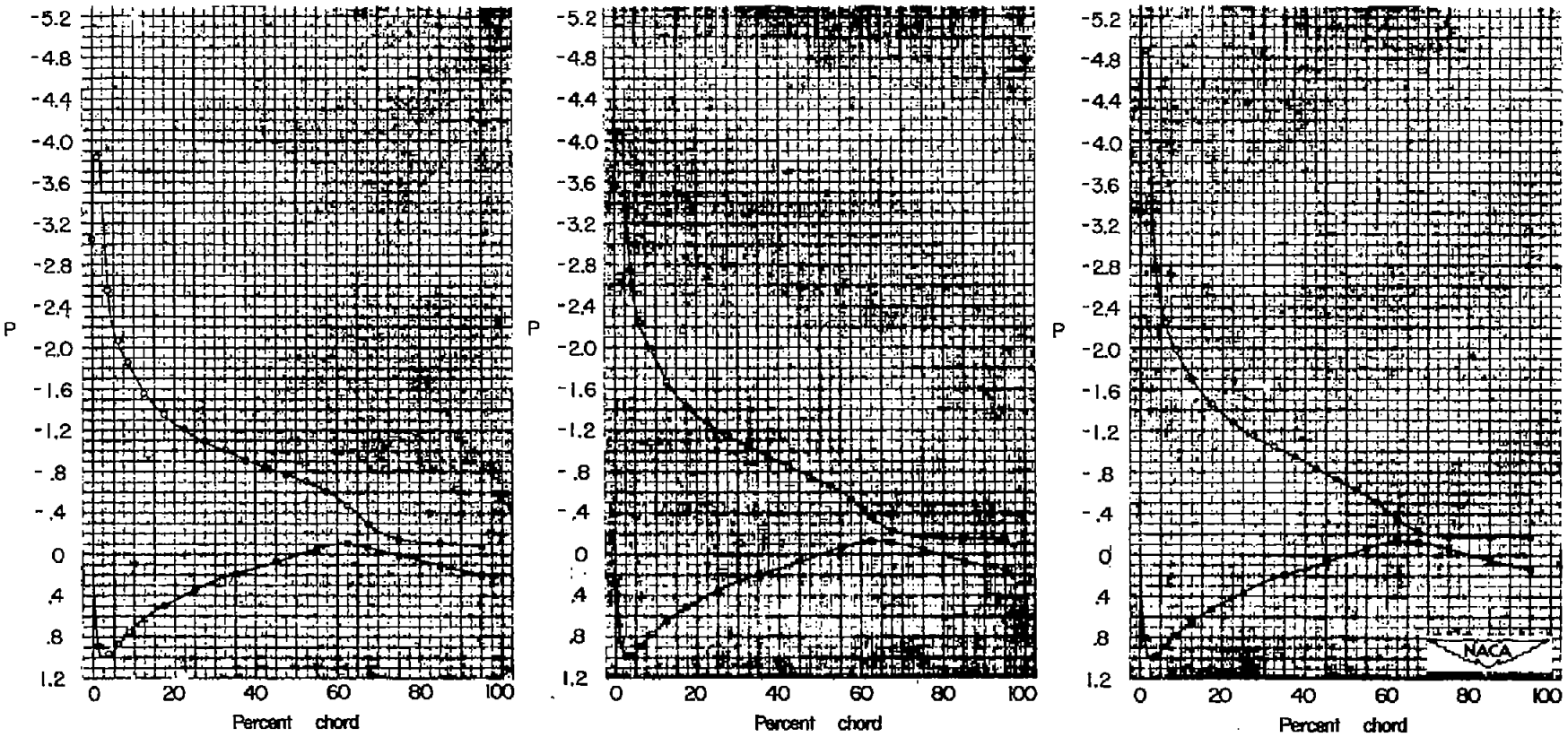
Station 5



Station 6

(c) Concluded. $\alpha = 14.17^{\circ}$.

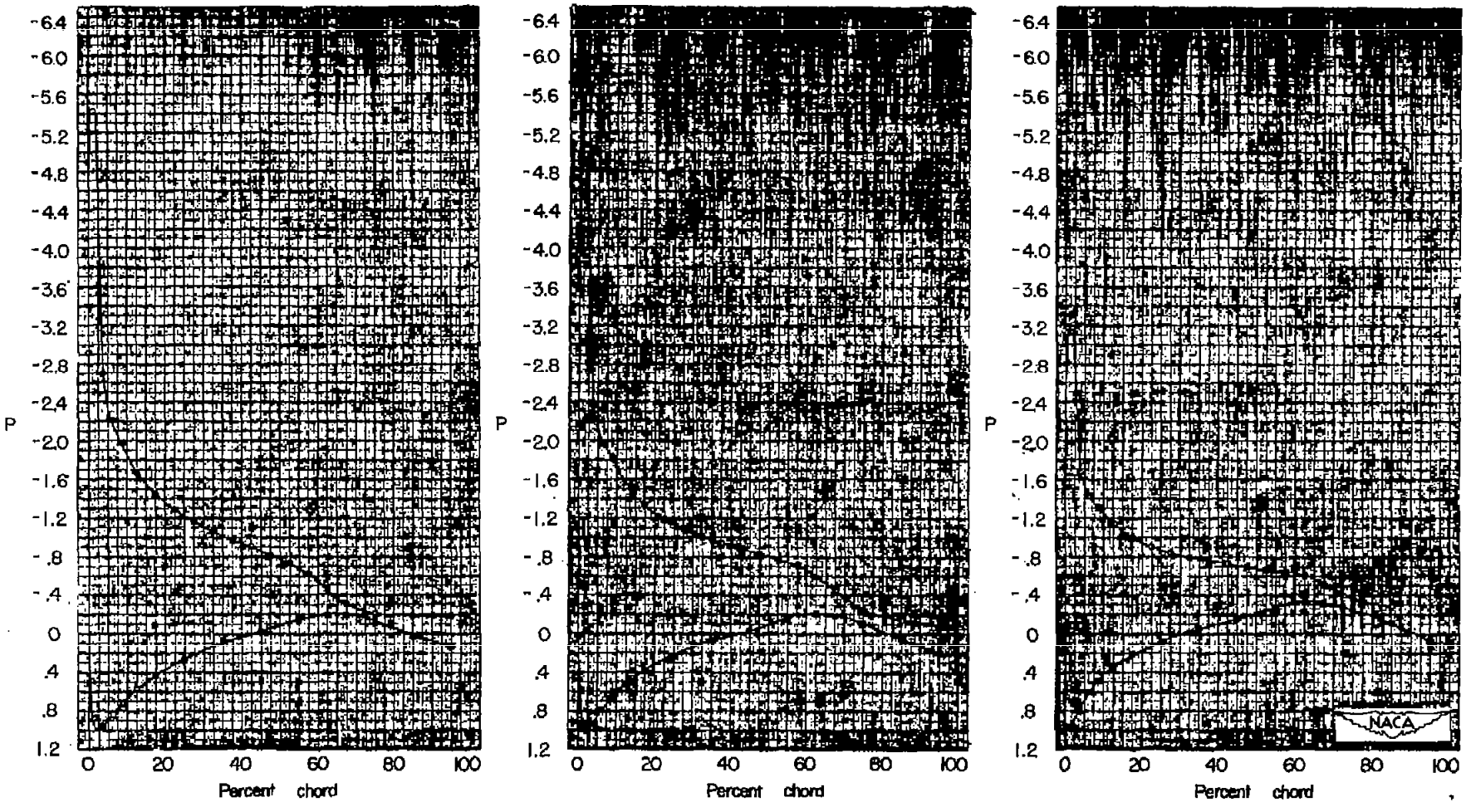
Figure 6.- Continued. $M = 0.30$.



Station 1 Station 2 Station 3

(d) $\alpha = 15.22^{\circ}$.

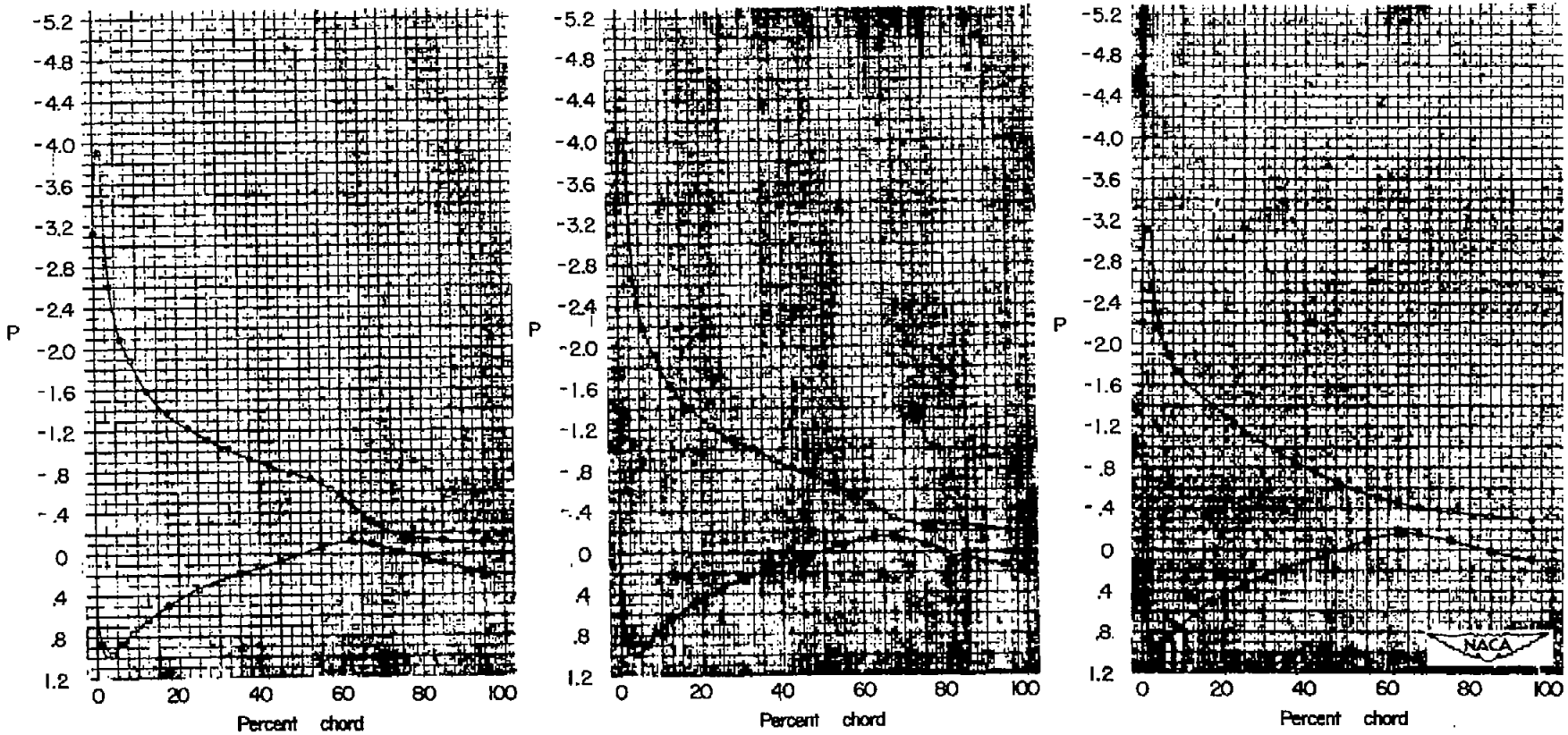
Figure 6.- Continued. M = 0.30.



Station 4 Station 5 Station 6

(d) Concluded. $\alpha = 15.22^{\circ}$.

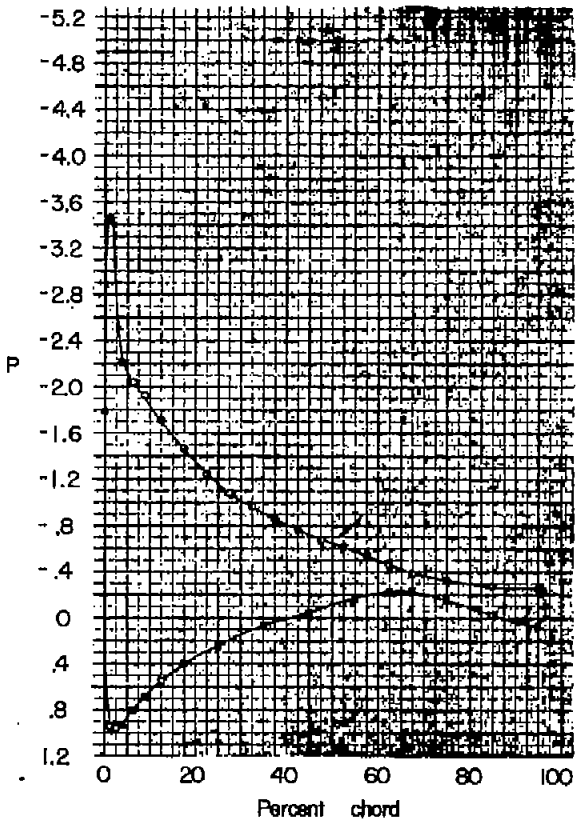
Figure 6.- Continued. M = 0.30.



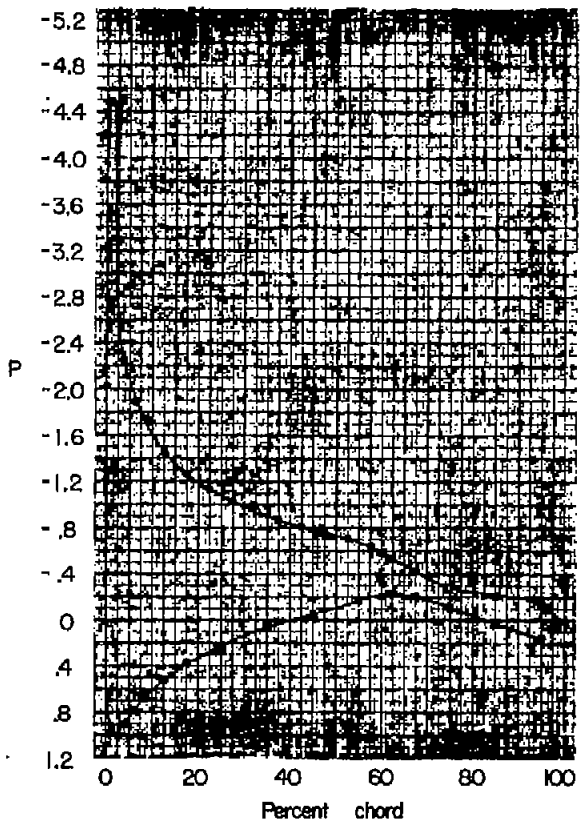
Station 1 Station 2 Station 3

(e) $\alpha = 15.74^{\circ}$.

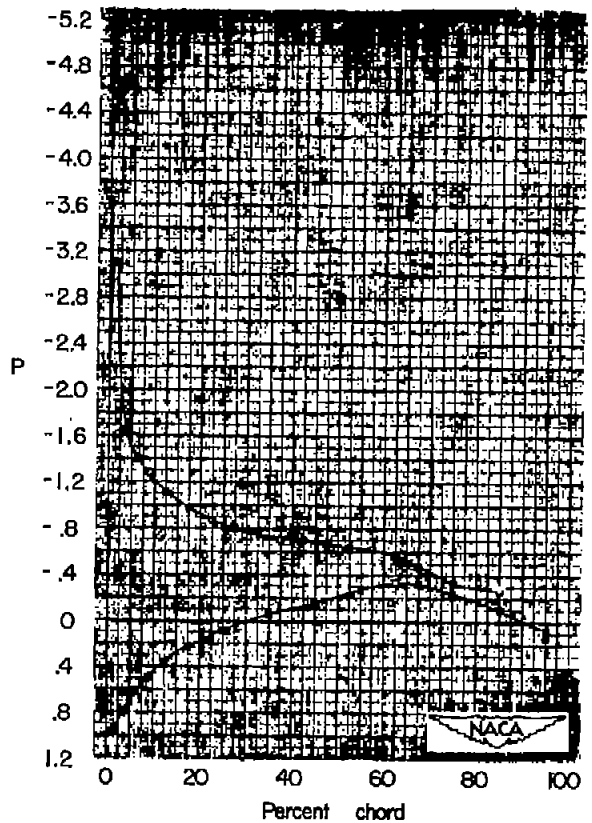
Figure 6.- Continued. M = 0.30.



Station 4



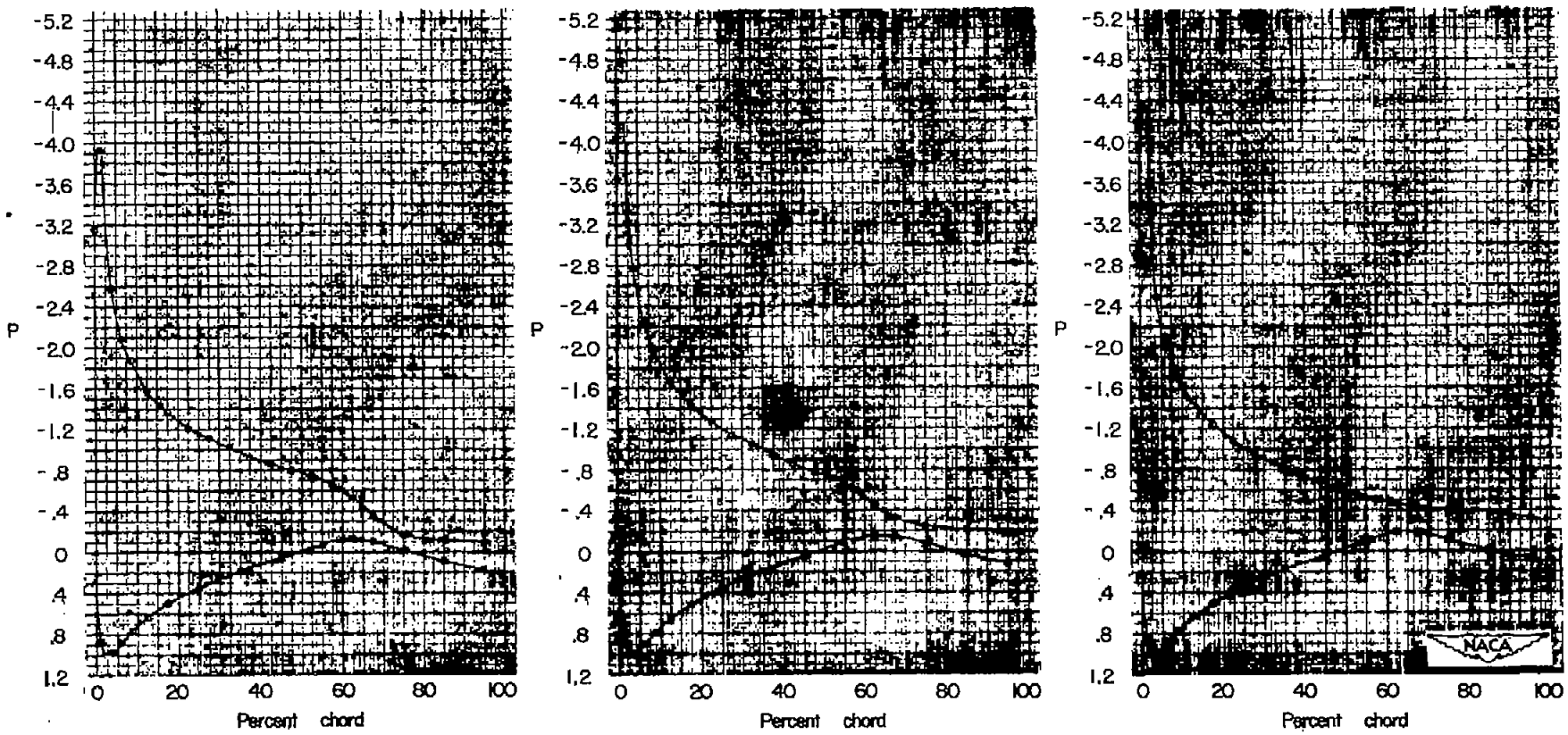
Station 5



Station 6

(e) Concluded. $\alpha = 15.74^{\circ}$.

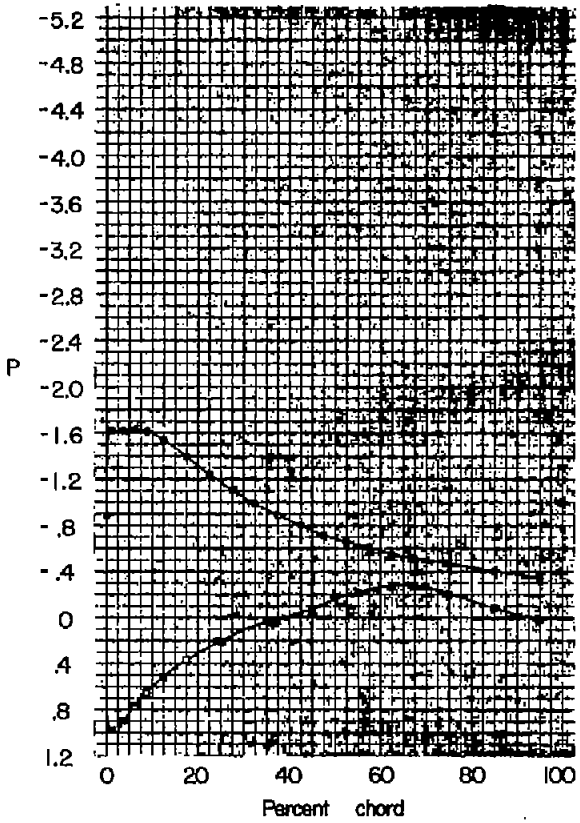
Figure 6.- Continued. $M = 0.30$.



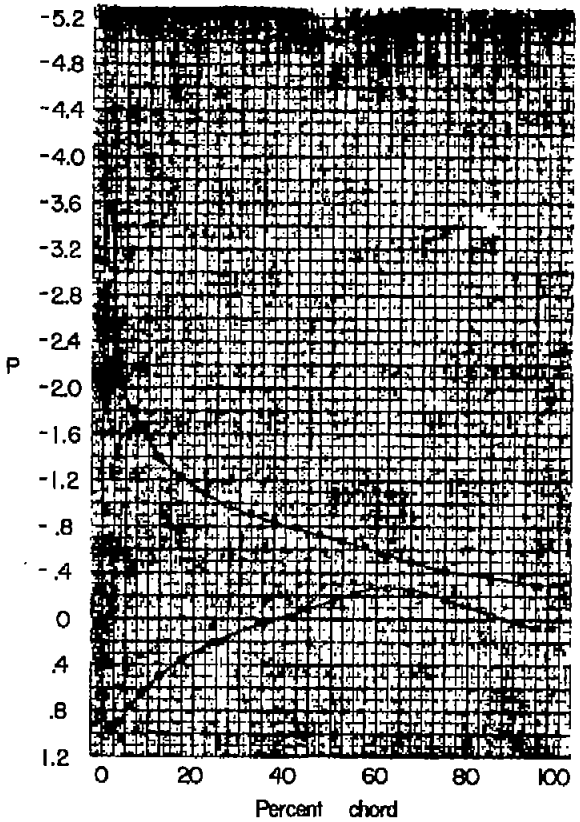
Station 1 Station 2 Station 3

(f) $\alpha = 16.27^{\circ}$.

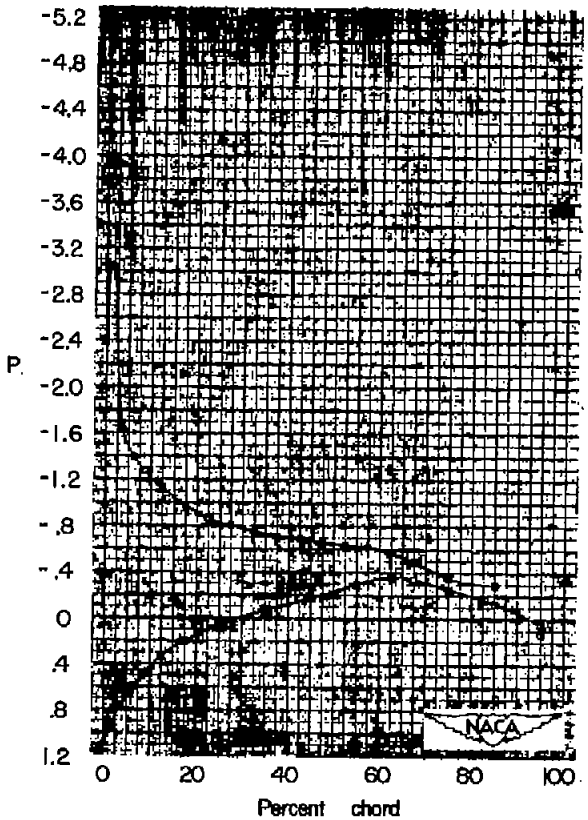
Figure 6.- Continued. M = 0.30.



Station 4



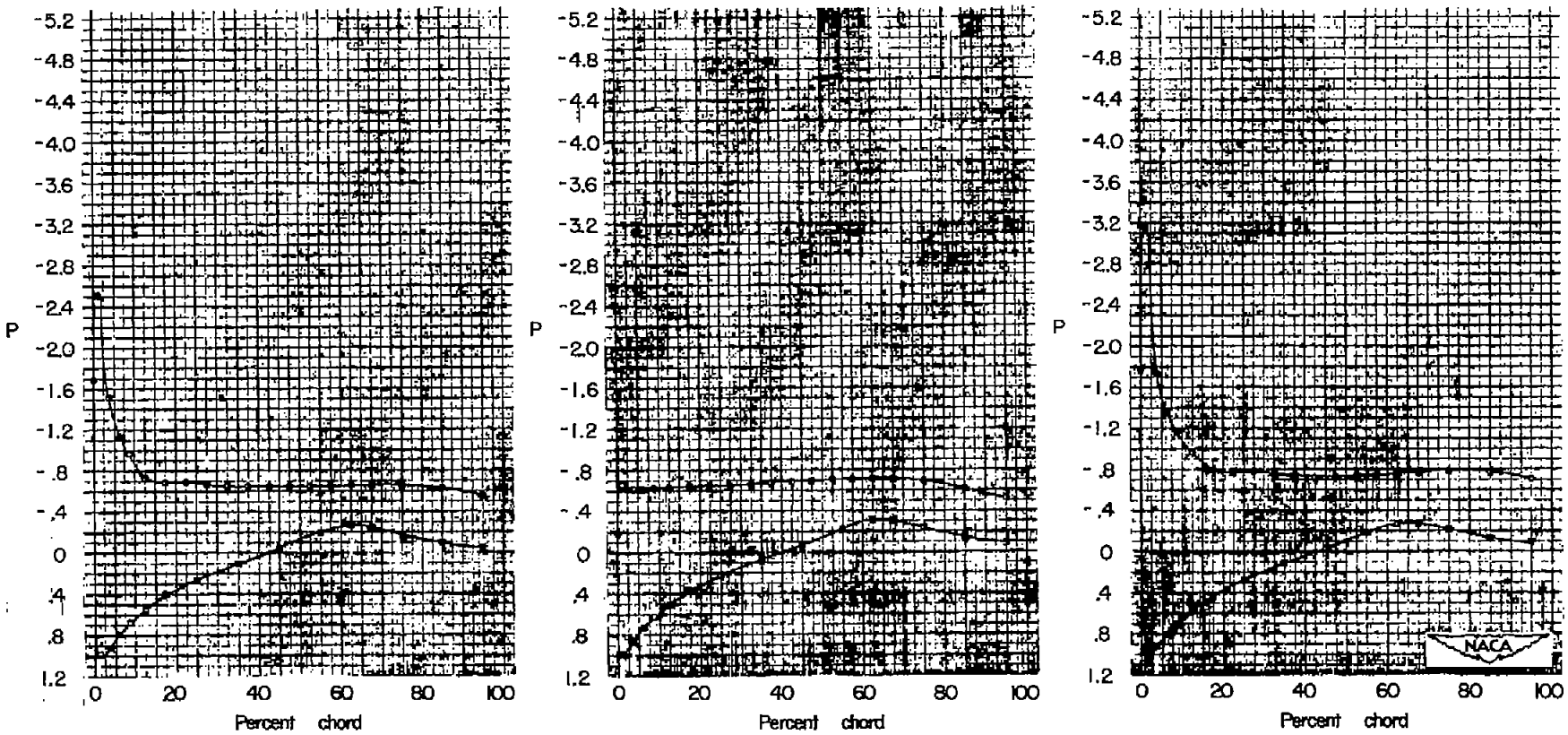
Station 5



Station 6

(f) Concluded. $\alpha = 16.27^{\circ}$.

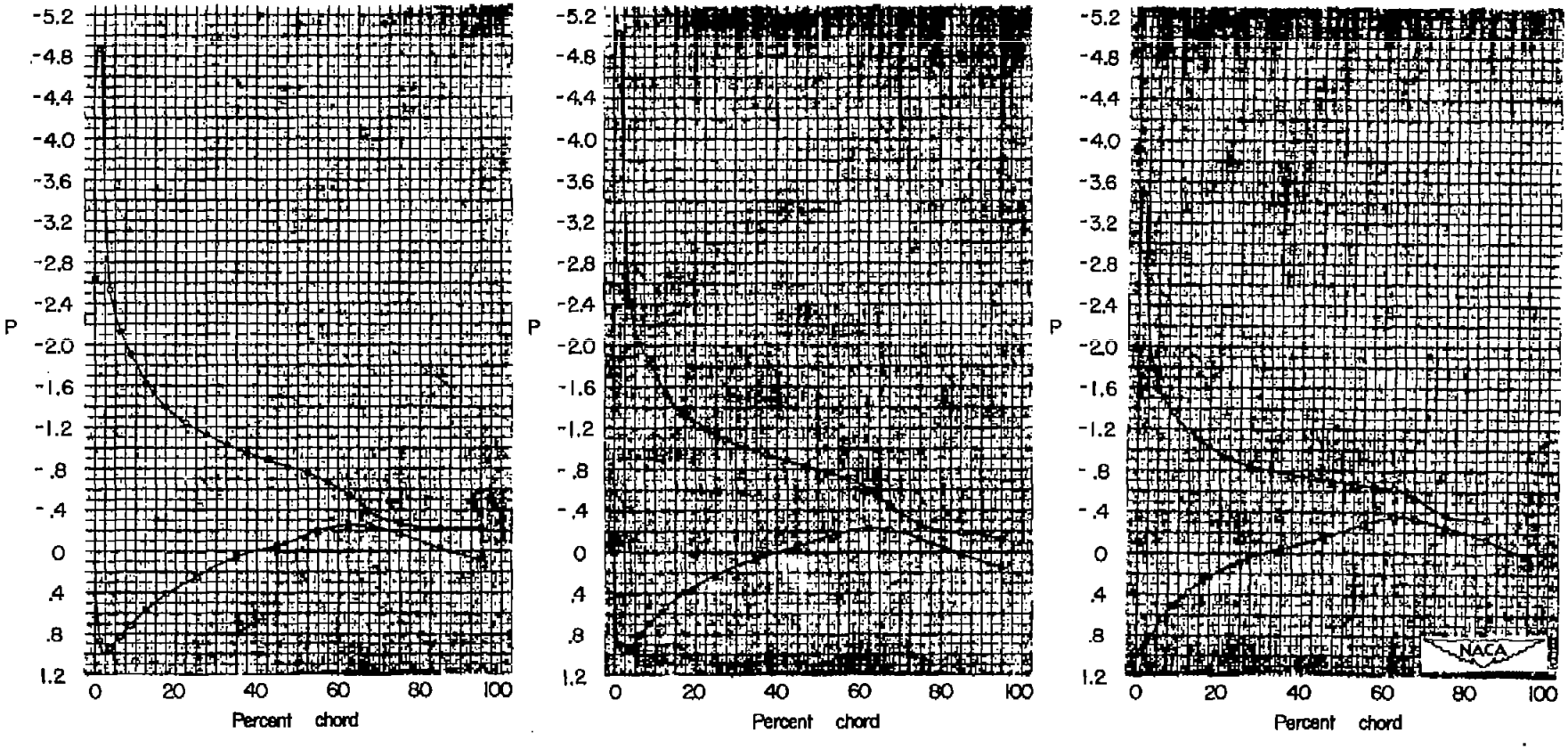
Figure 6.- Continued. M = 0.30.



Station 1 Station 2 Station 3

(g) $\alpha = 17.30^{\circ}$.

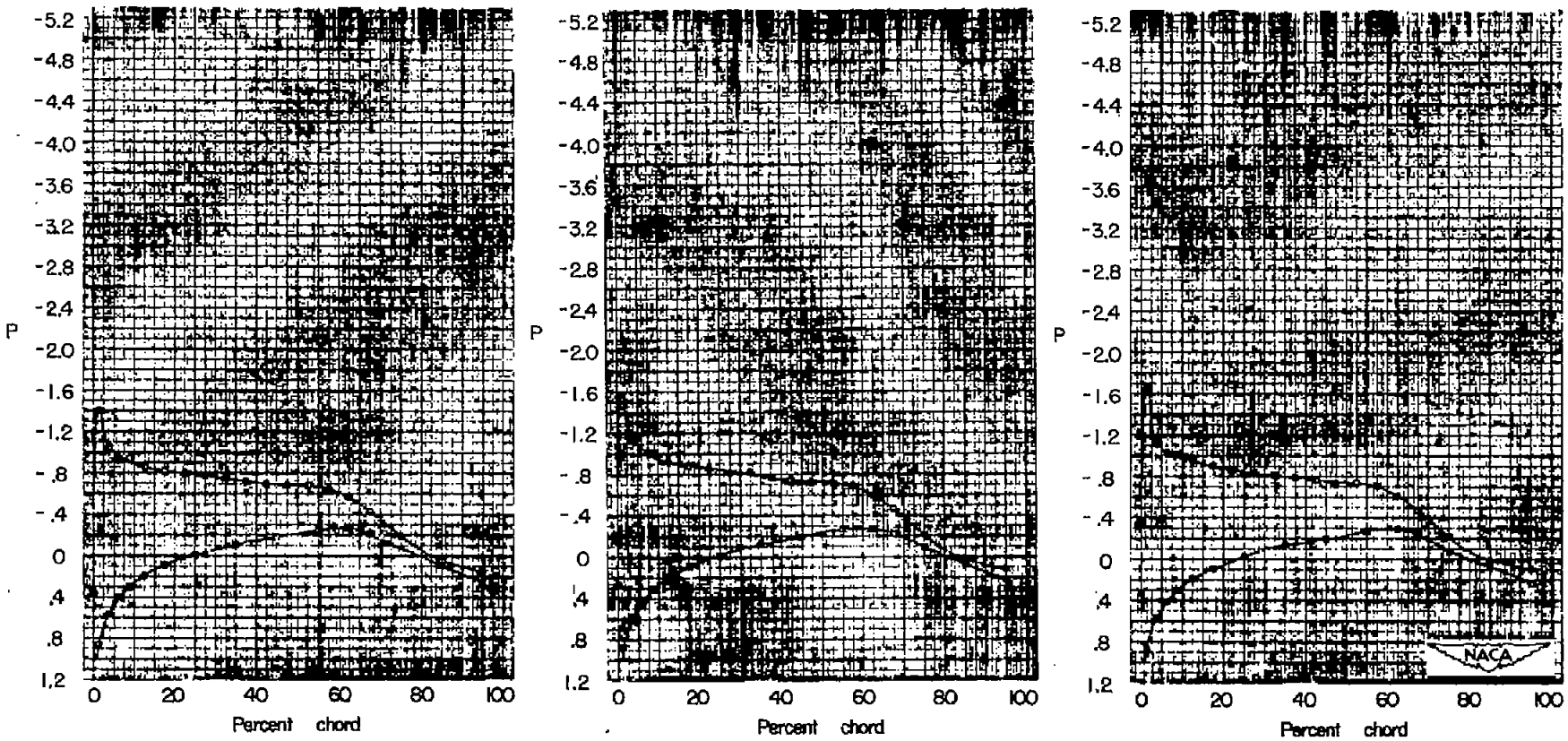
Figure 6.- Continued. M = 0.30.



Station 4 Station 5 Station 6

(g) Concluded. $\alpha = 17.30^{\circ}$.

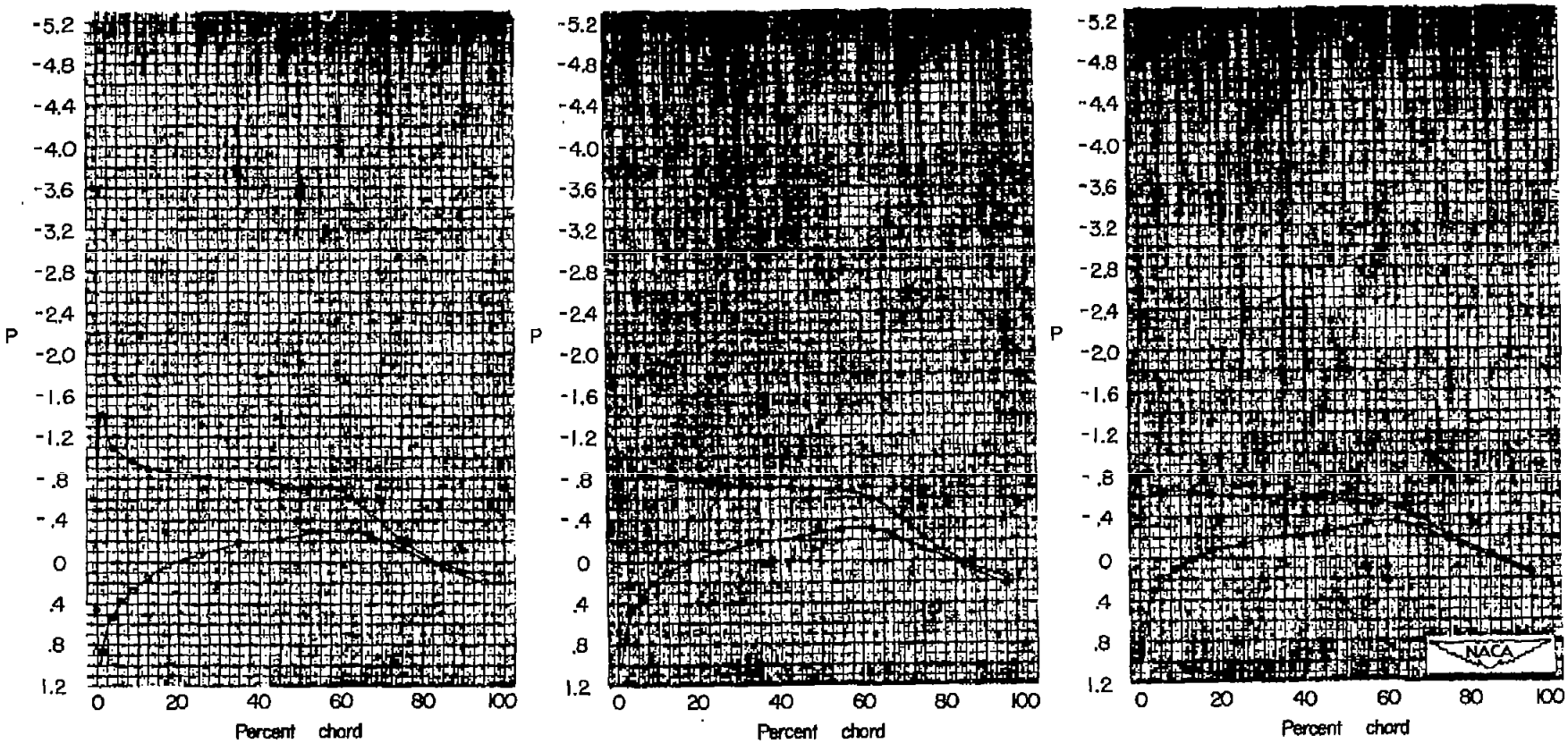
Figure 6.- Concluded. M = 0.30.



Station 1 Station 2 Station 3

(a) $\alpha = 6.74^{\circ}$.

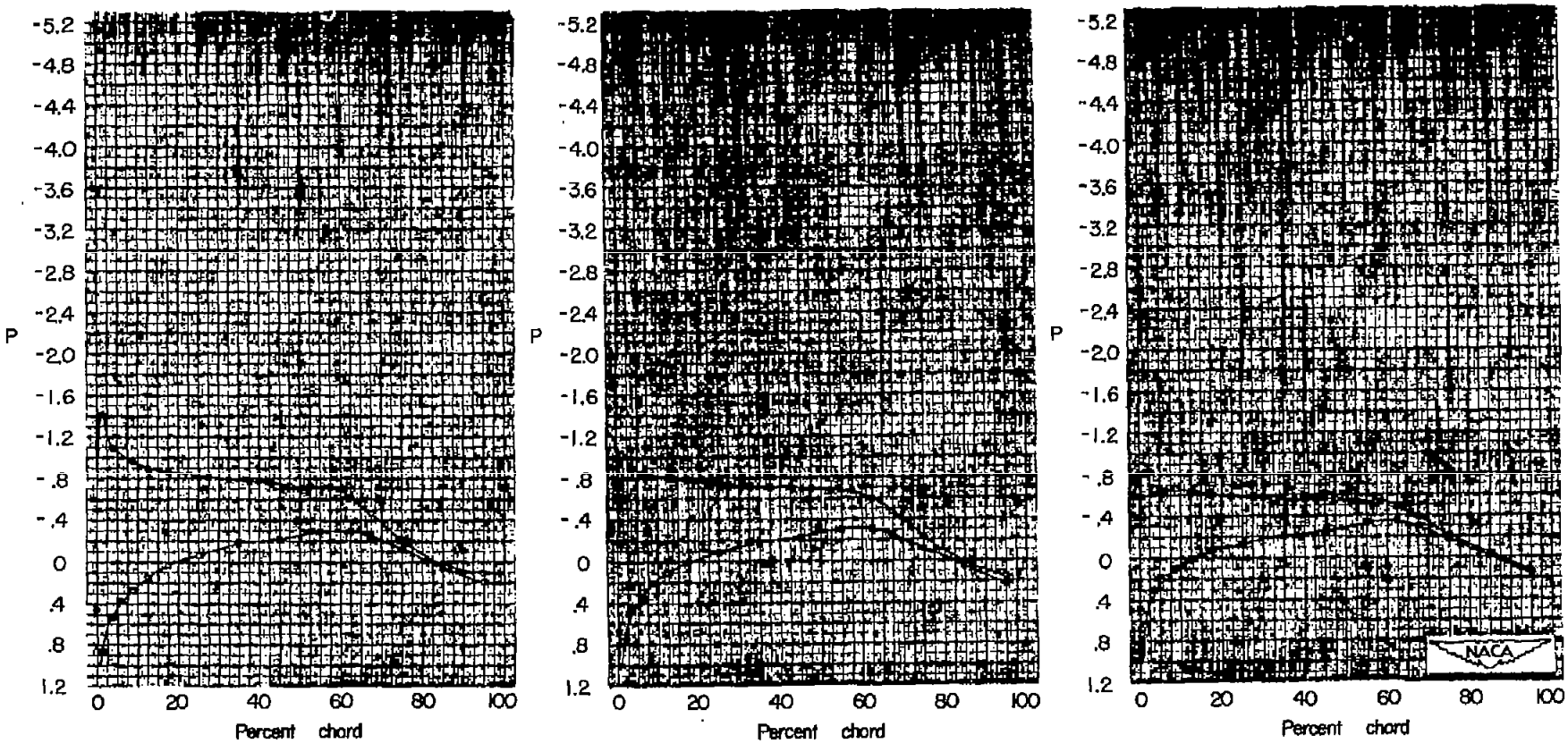
Figure 7.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.35.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = 6.74^{\circ}$.

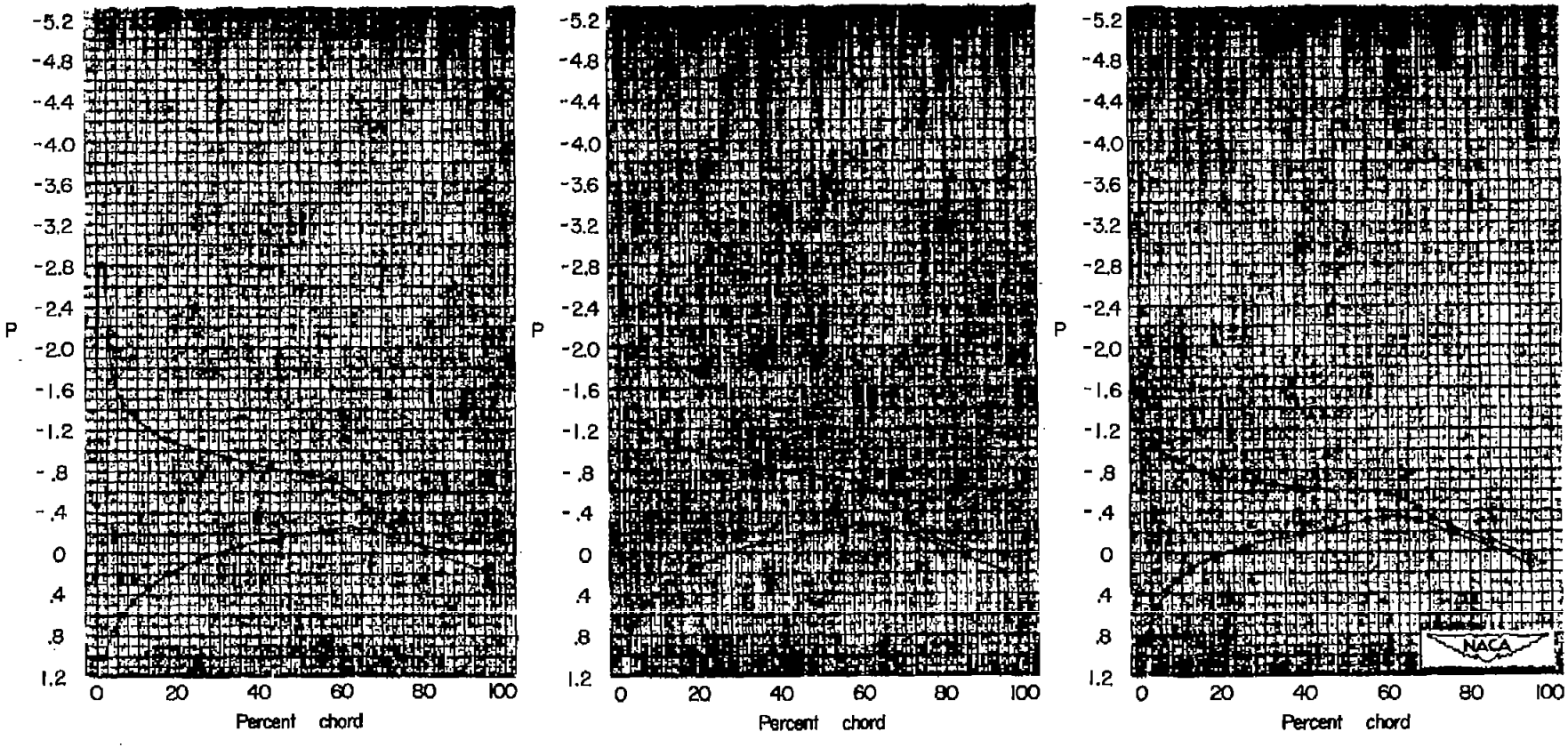
Figure 7.- Continued. M = 0.35.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = 6.74^{\circ}$.

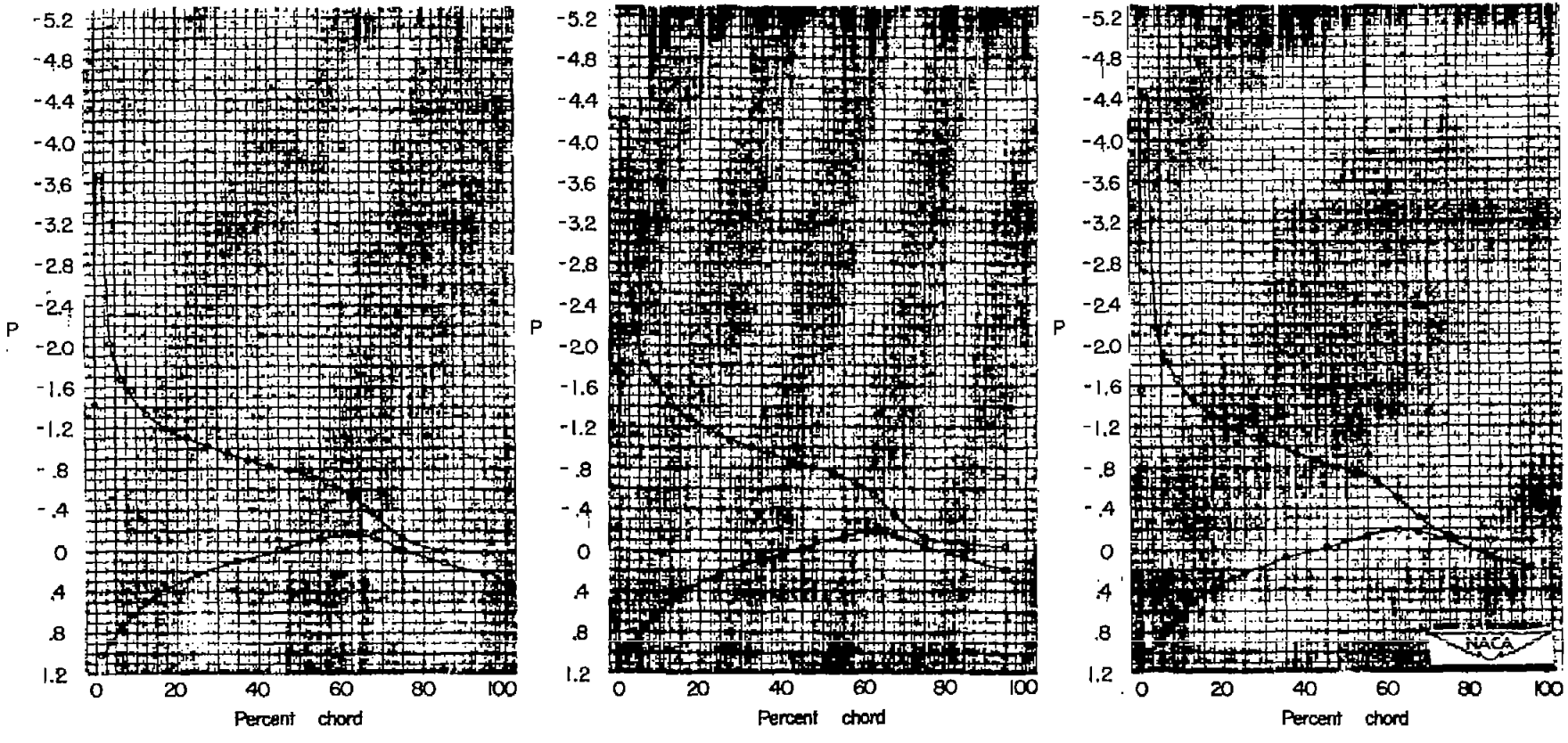
Figure 7.- Continued. M = 0.35.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = 9.98^{\circ}$.

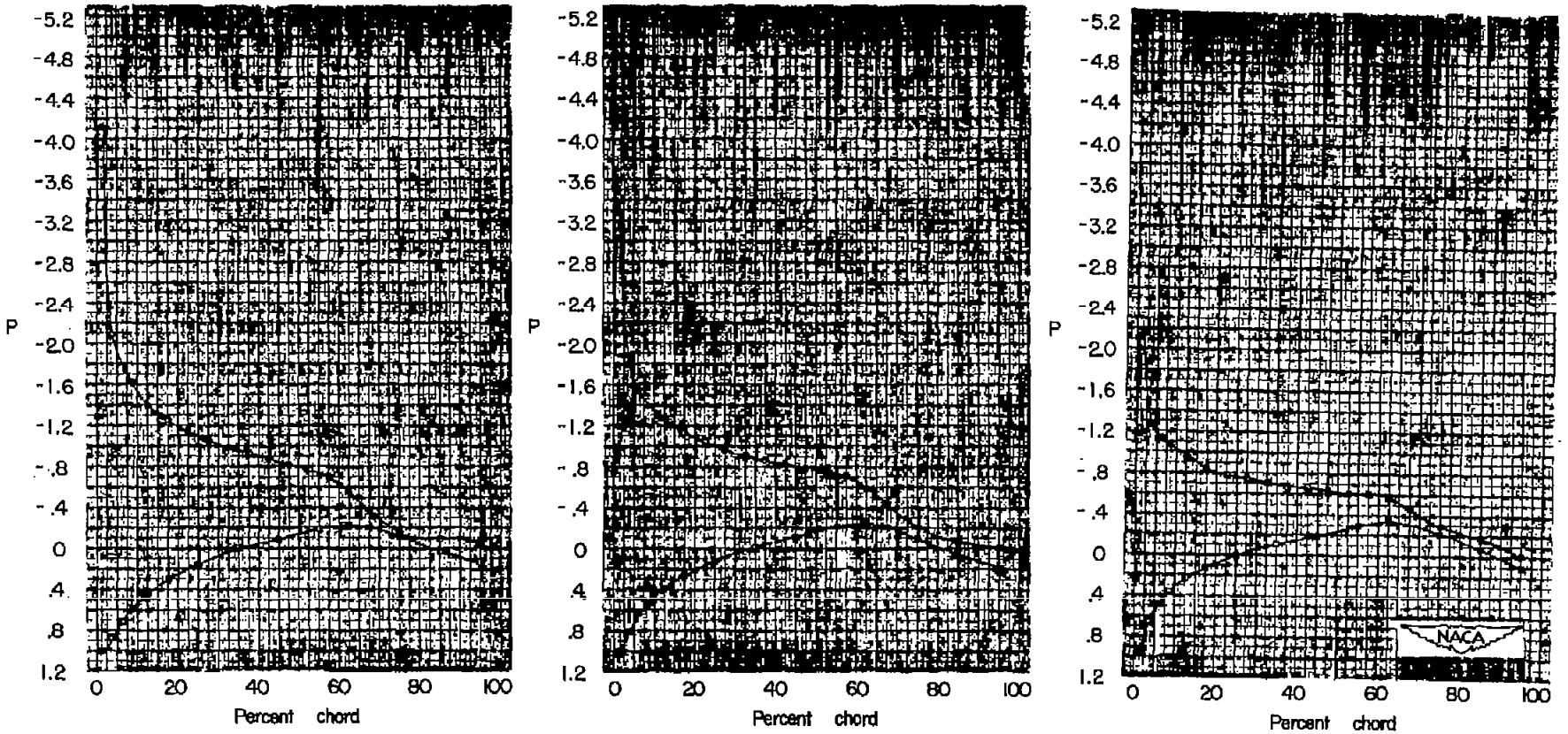
Figure 7.- Continued. M = 0.35.



Station 1 Station 2 Station 3

(c) $\alpha = 12.10^{\circ}$.

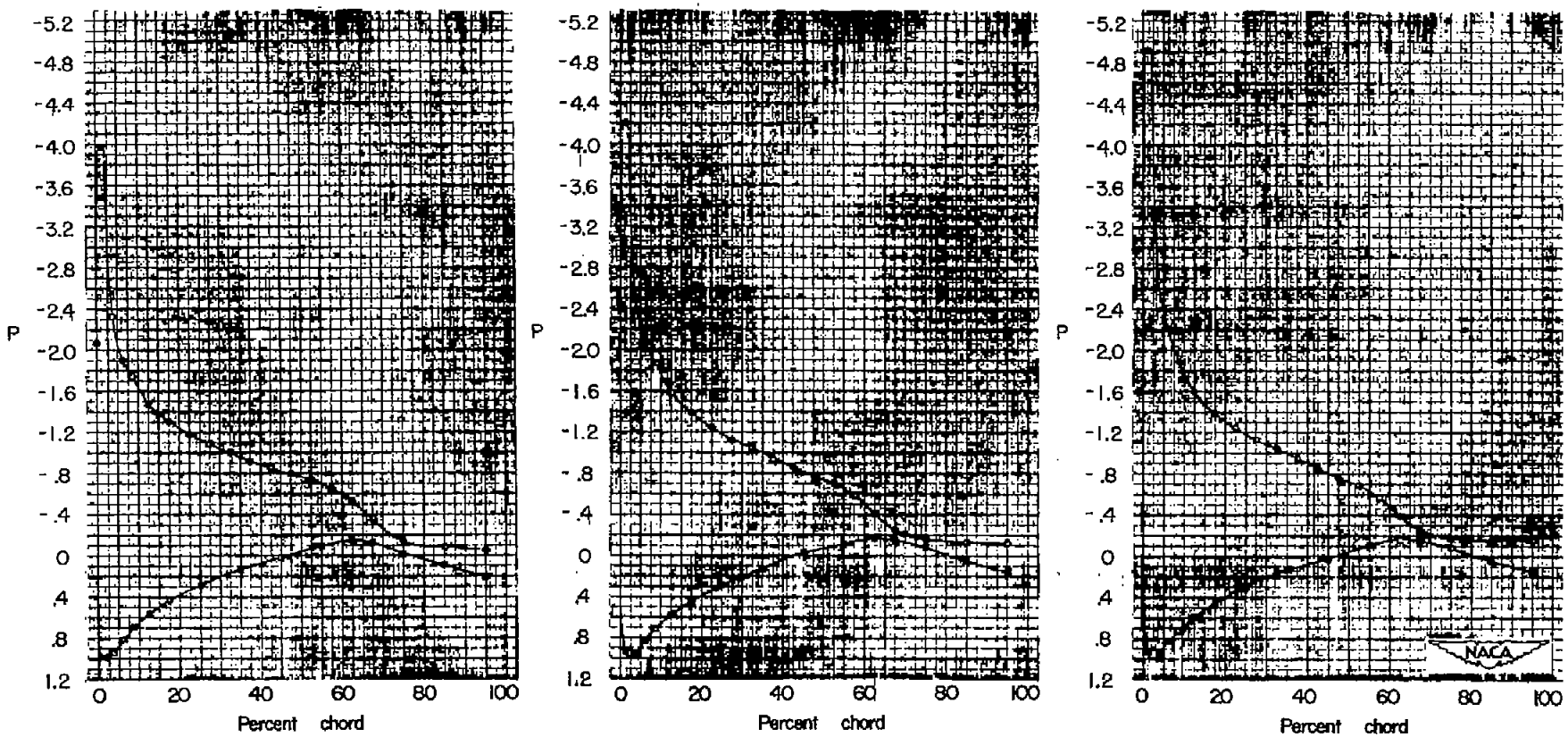
Figure 7.- Continued. M = 0.35.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 12.10^{\circ}$.

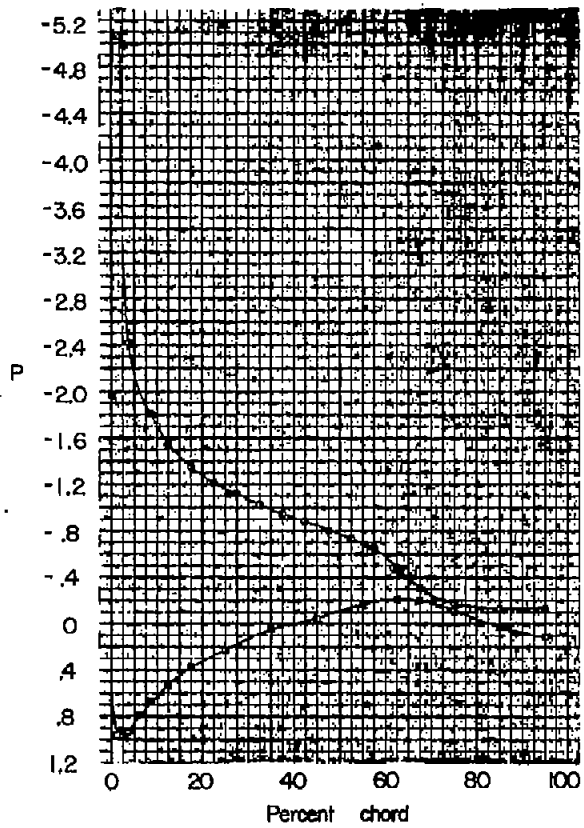
Figure 7.- Continued. M = 0.35.



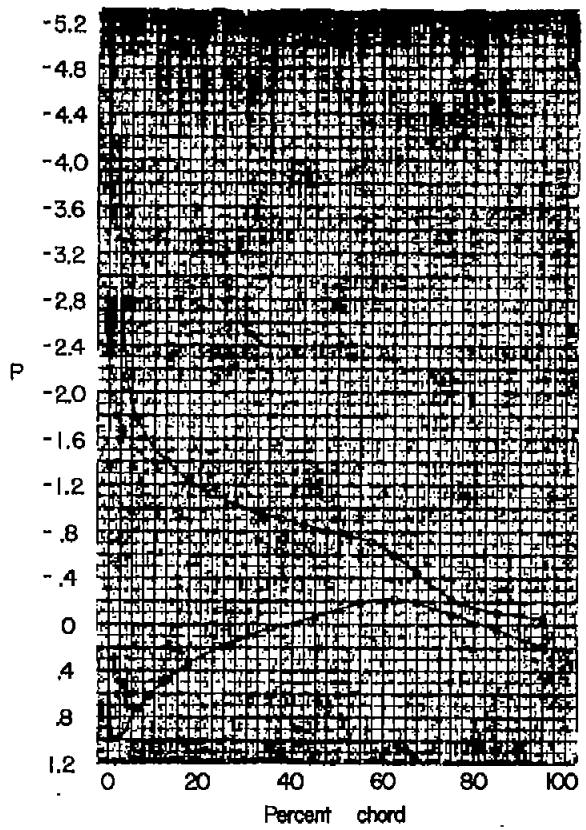
Station 1 Station 2 Station 3

(d) $\alpha = 13.75^{\circ}$.

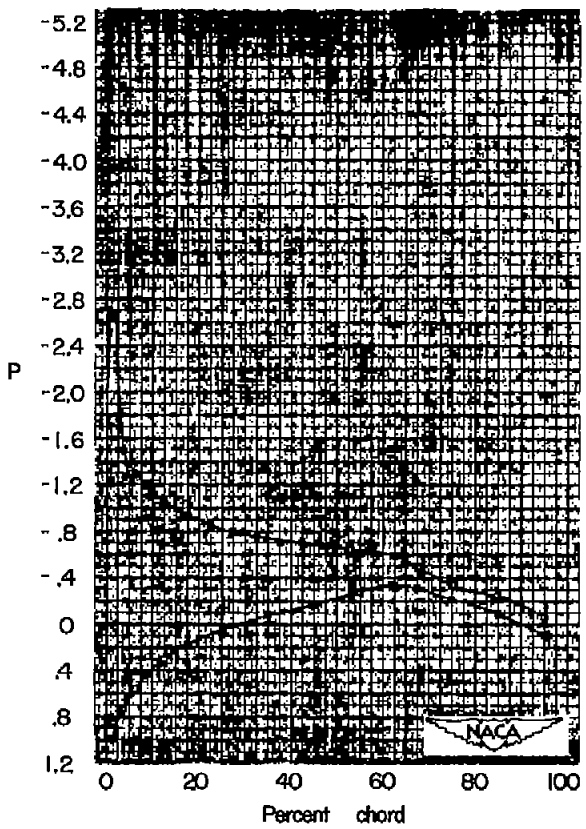
Figure 7.- Continued. M = 0.35.



Station 4



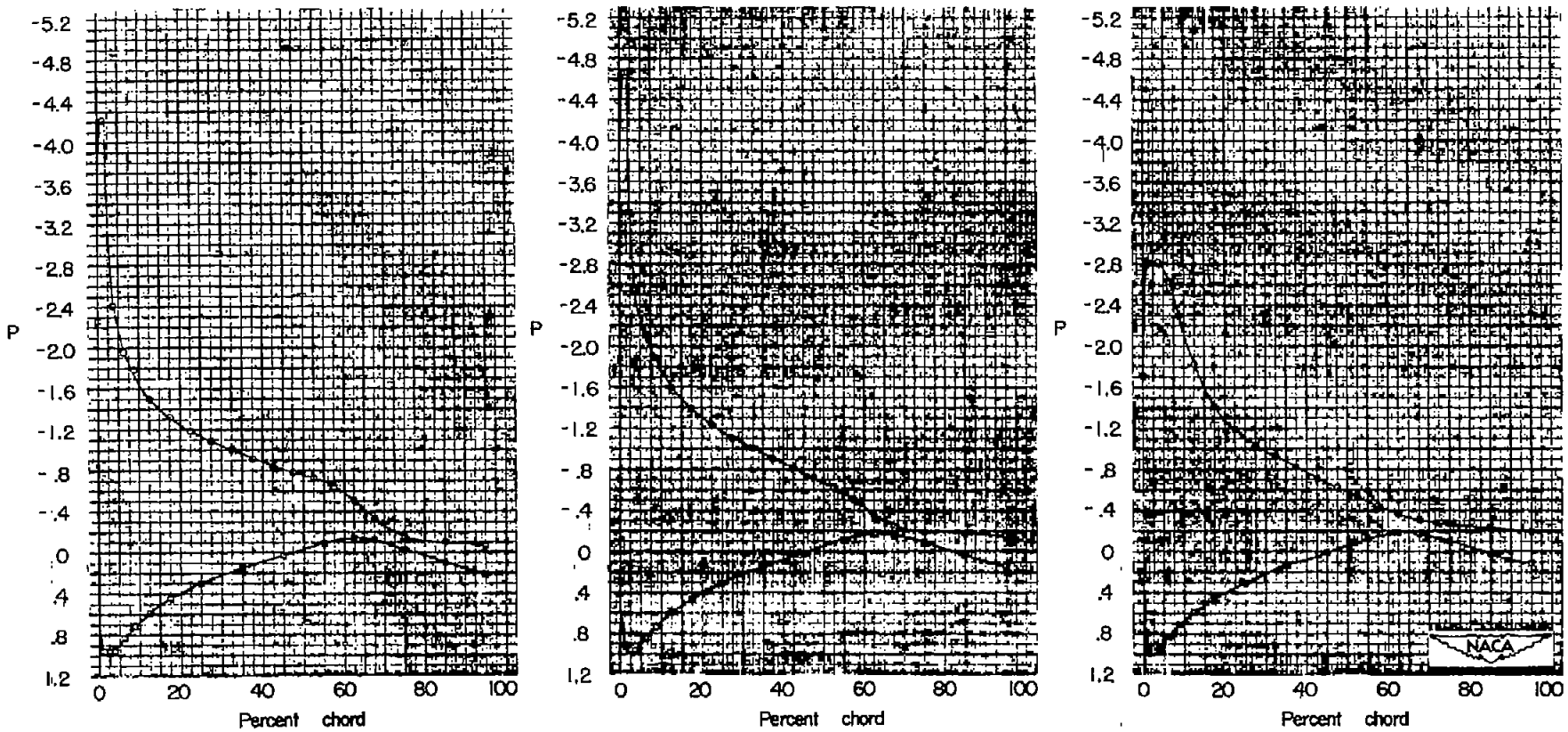
Station 5



Station 6

(d) Concluded. $\alpha = 13.75^{\circ}$.

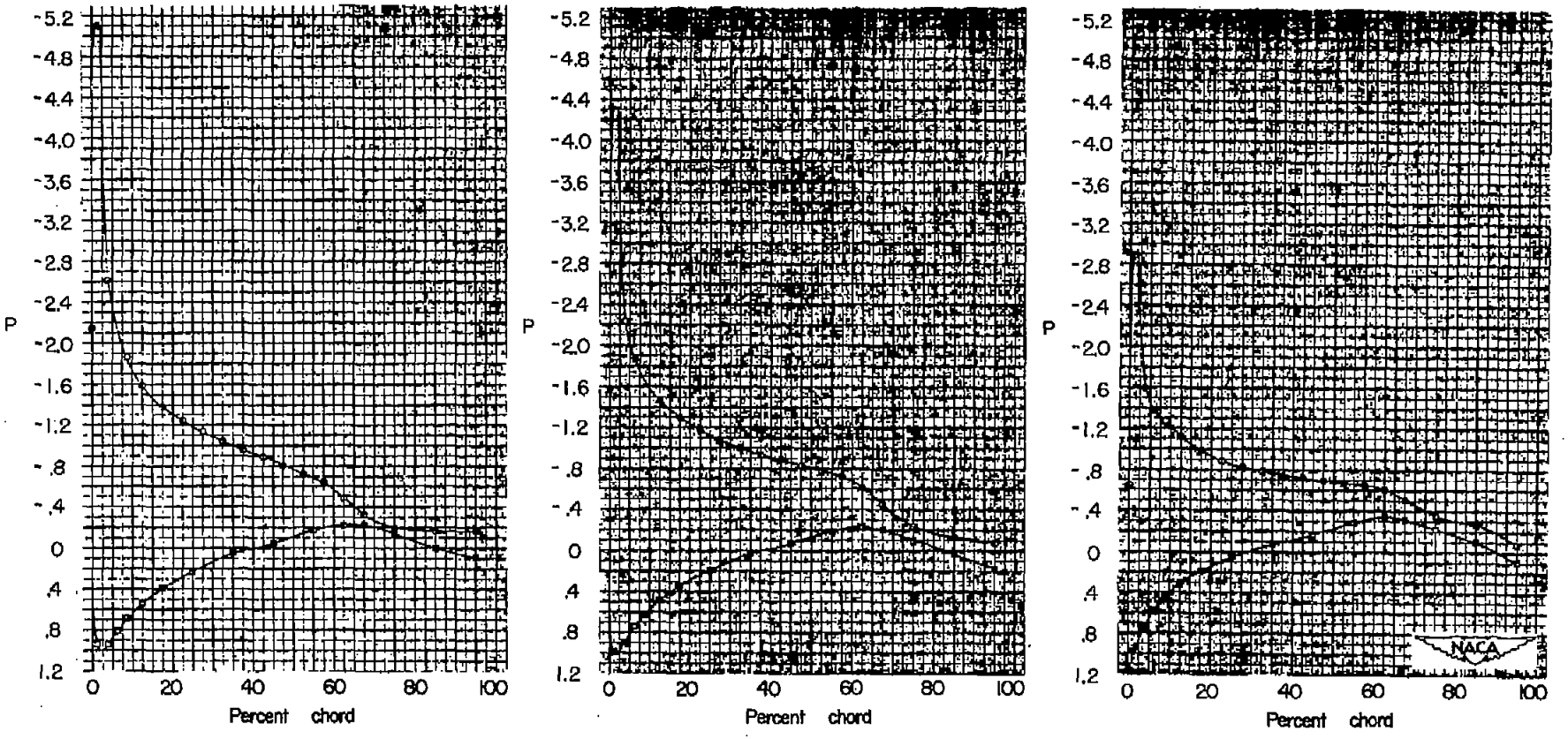
Figure 7.- Continued. $M = 0.35$.



Station 1 Station 2 Station 3

(e) $\alpha = 14.38^{\circ}$.

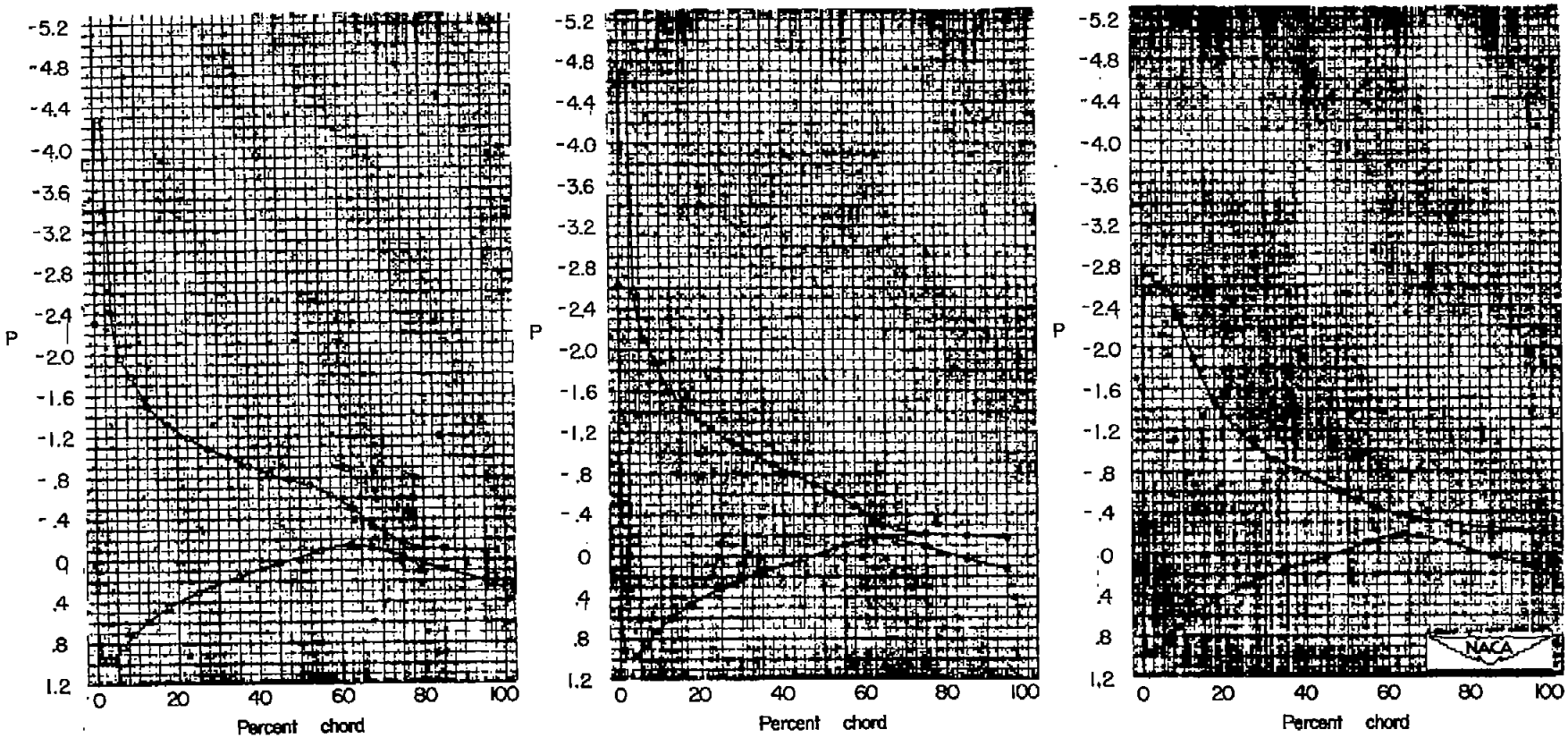
Figure 7.- Continued. M = 0.35.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 14.38^{\circ}$.

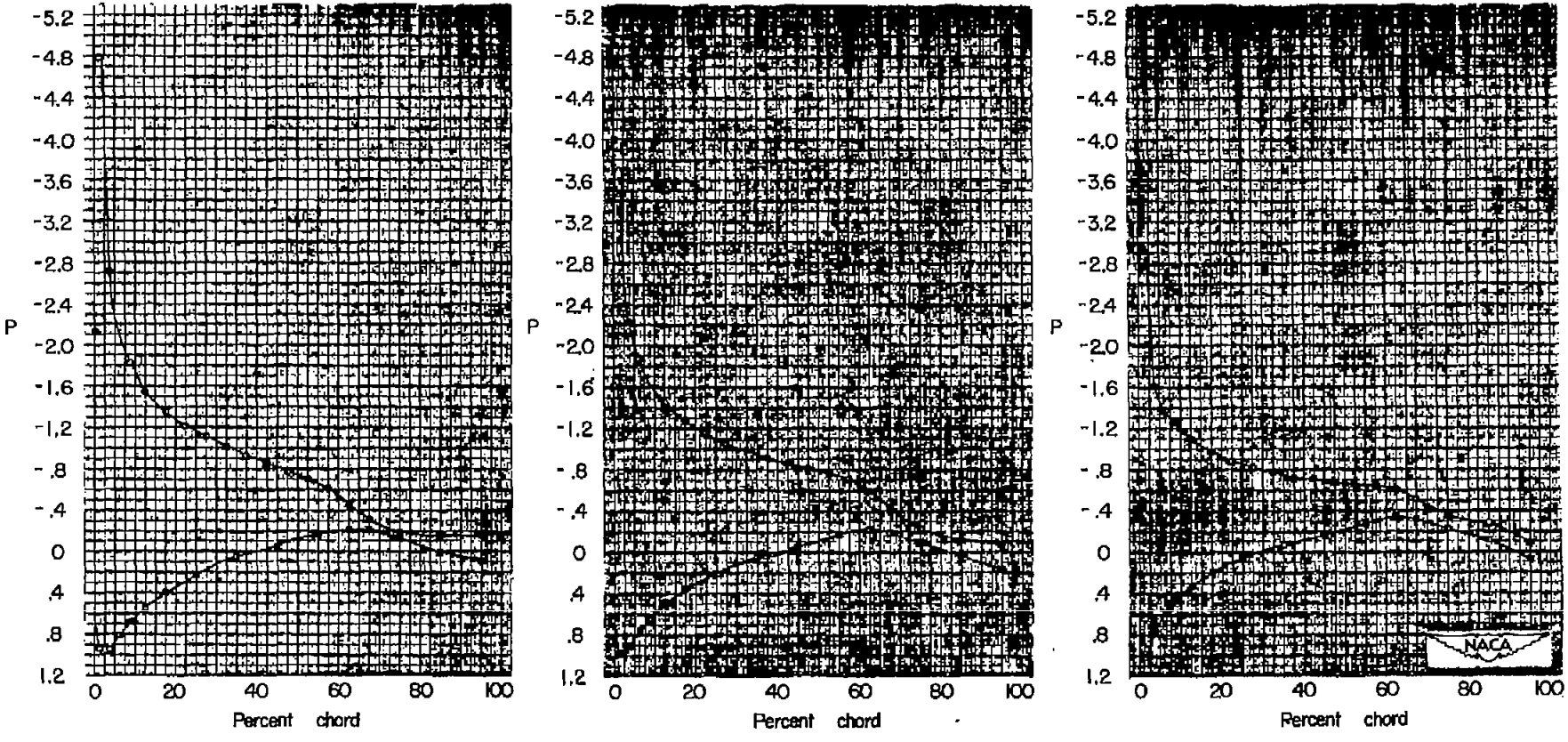
Figure 7.- Continued. M = 0.35.



Station 1 Station 2 Station 3

(f) $\alpha = 14.59^{\circ}$.

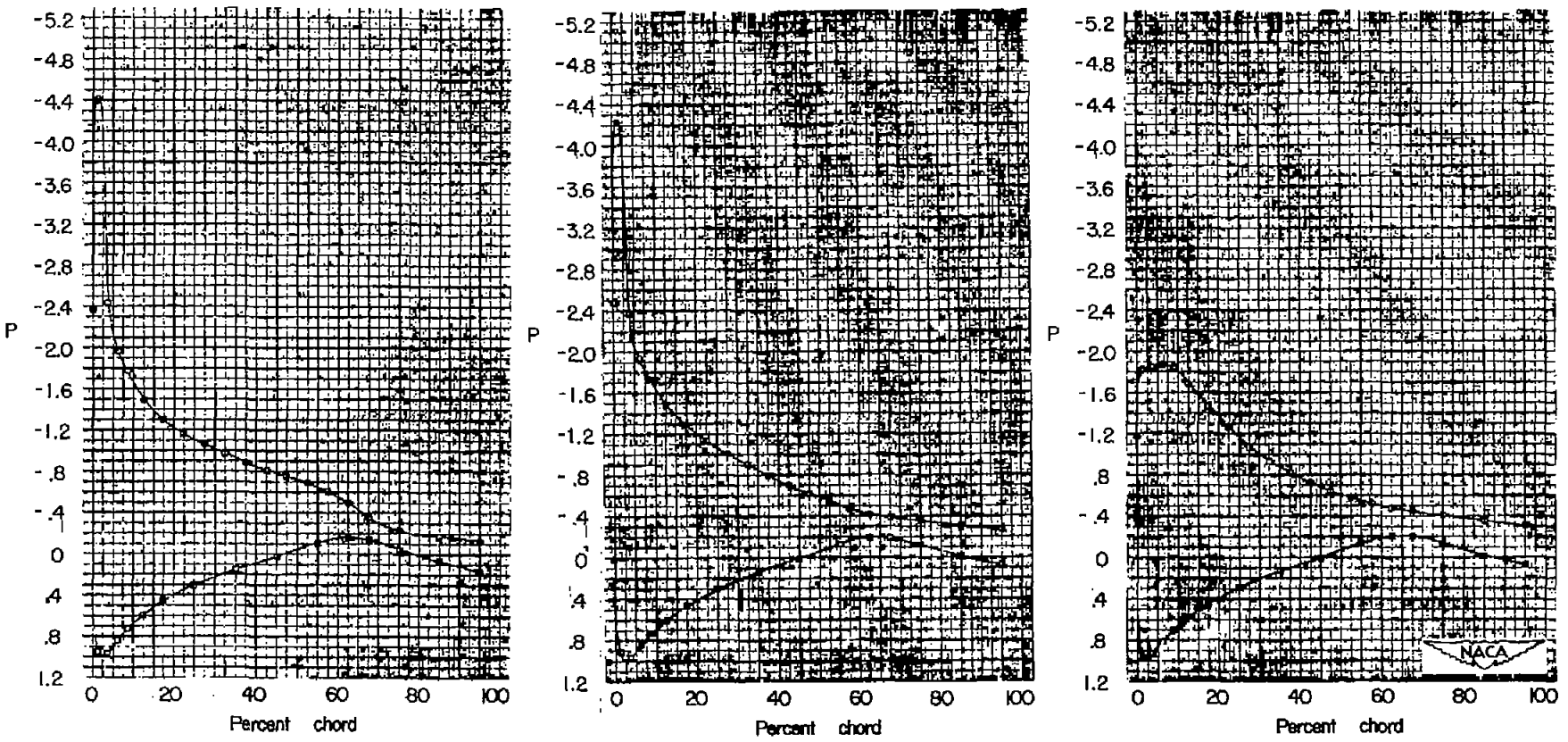
Figure 7.- Continued. M = 0.35.



Station 4 Station 5 Station 6

(f) Concluded. $\alpha = 14.59^{\circ}$.

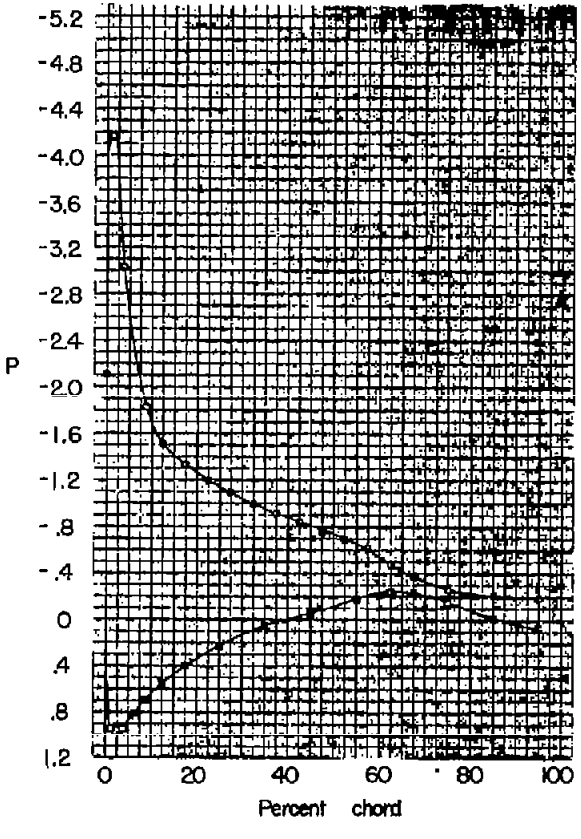
Figure 7.- Continued. $M = 0.35$.



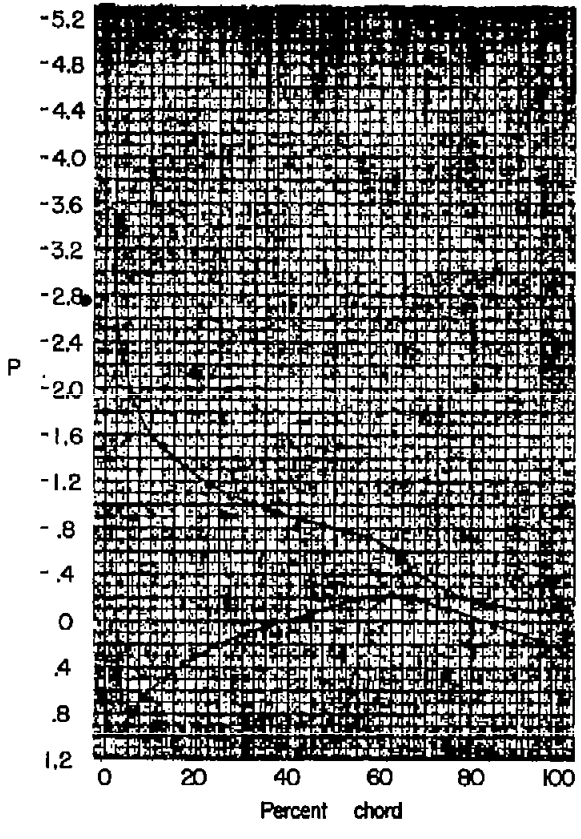
Station 1 Station 2 Station 3

(g) $\alpha = 15.22^{\circ}$.

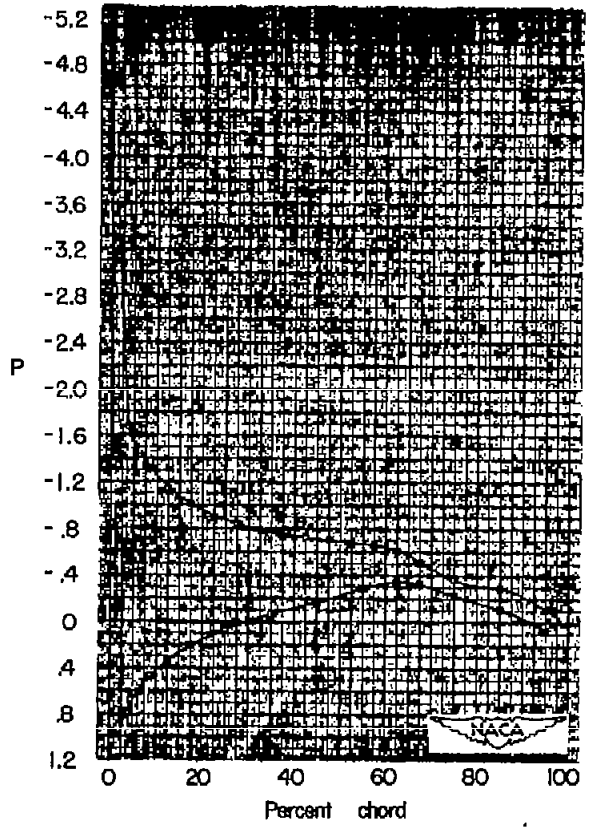
Figure 7.- Continued. M = 0.35.



Station 4



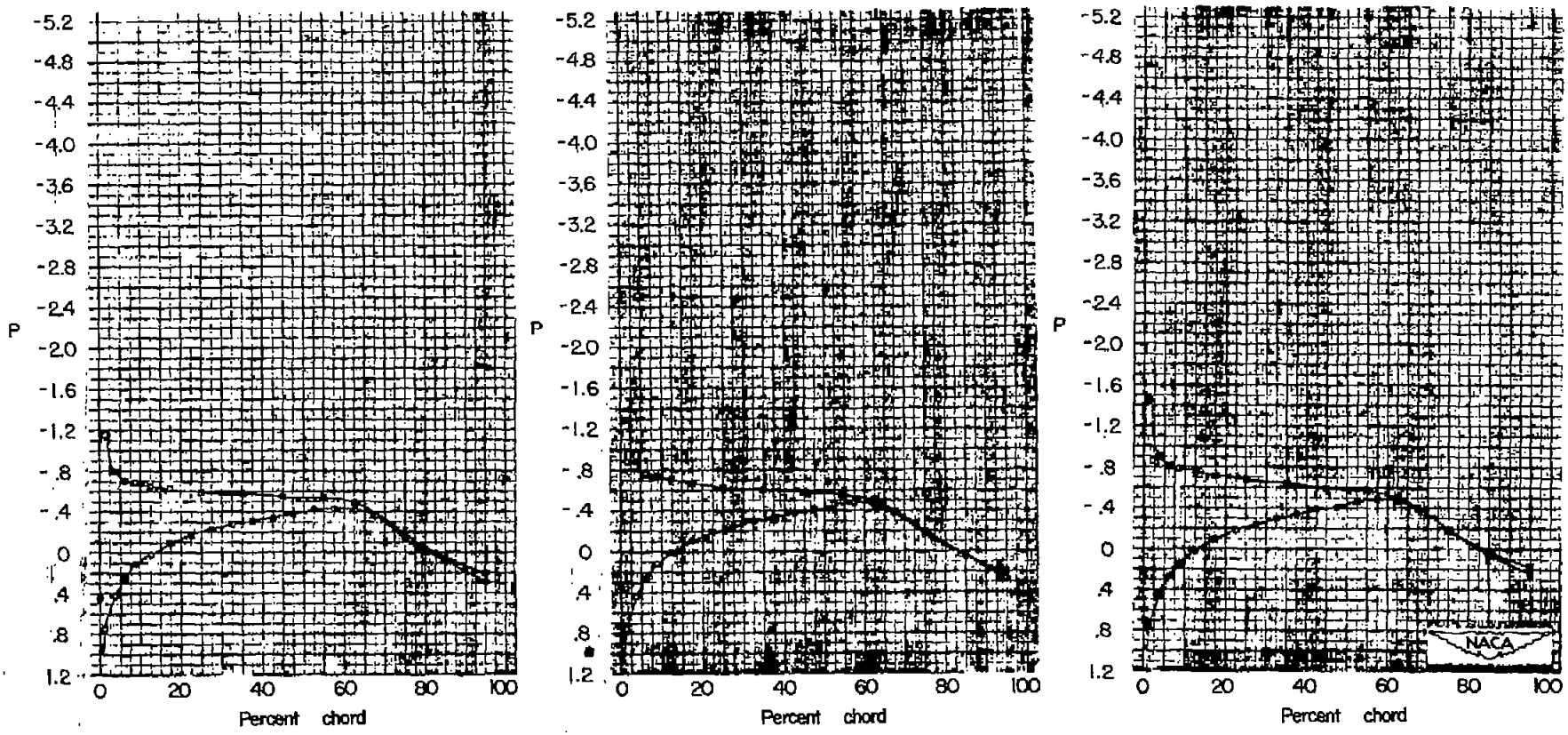
Station 5



Station 6

(g) Concluded. $\alpha = 15.22^{\circ}$.

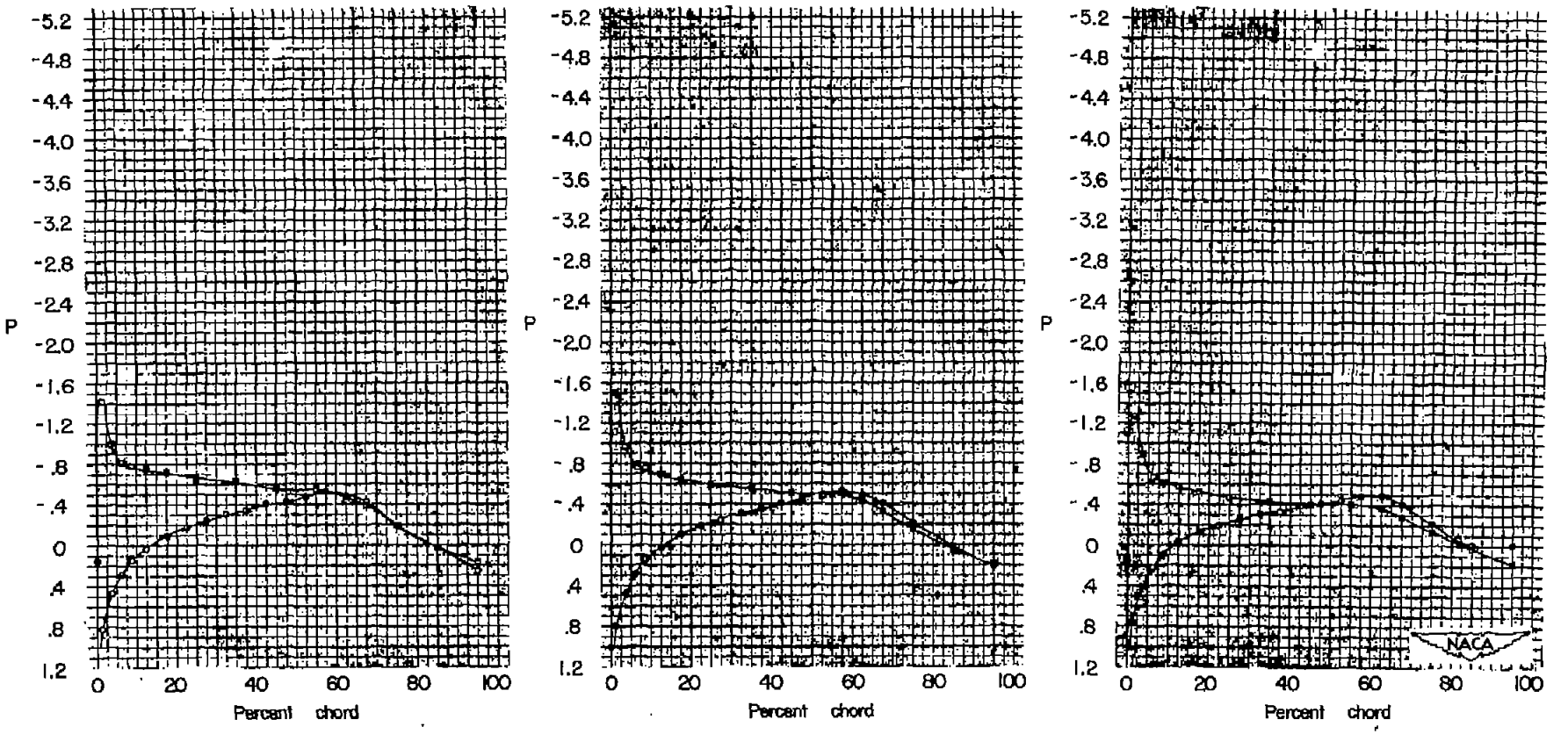
Figure 7.- Concluded. M = 0.35.



Station 1 Station 2 Station 3

(a) $\alpha = -4.13^{\circ}$.

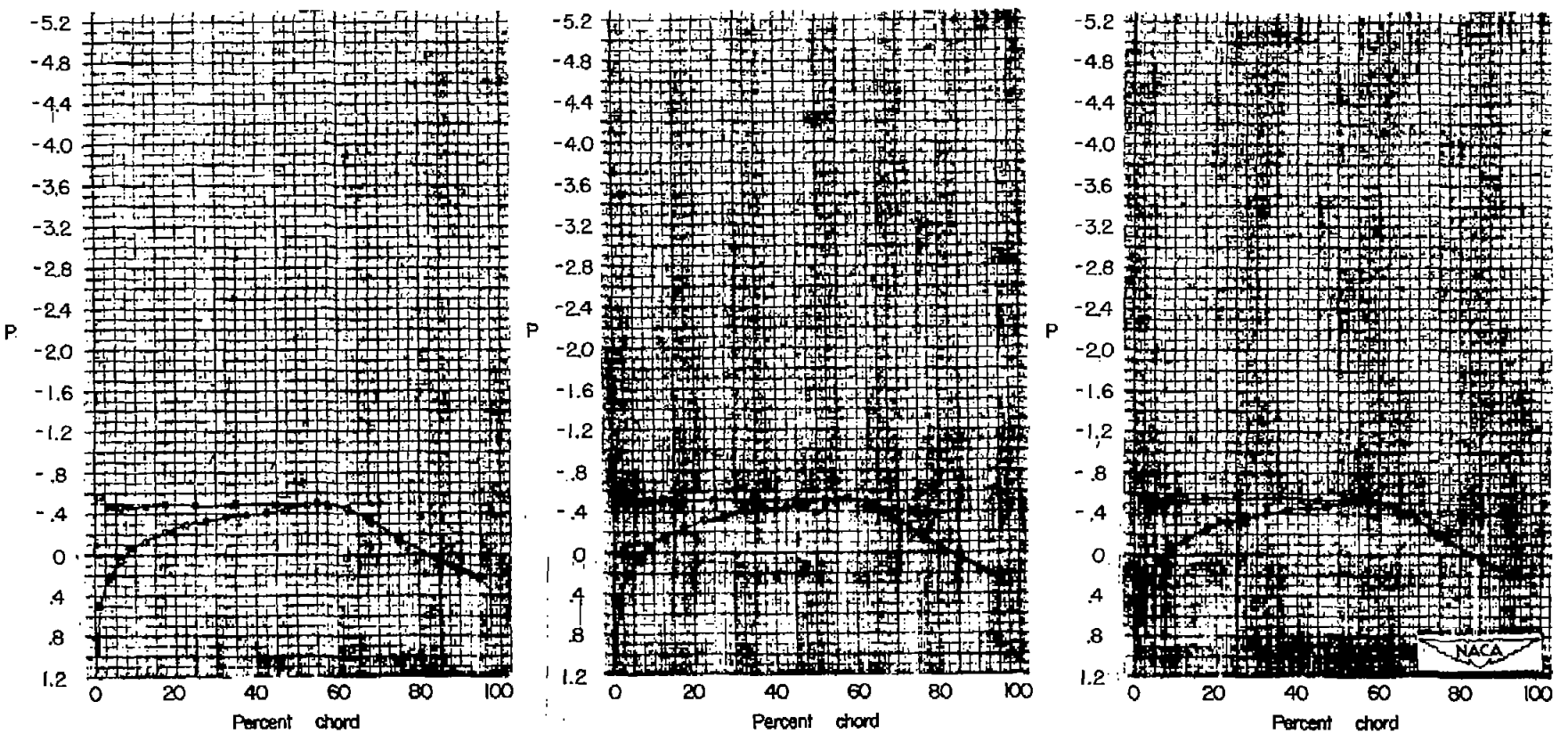
Figure 8.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.40.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = -4.13^{\circ}$.

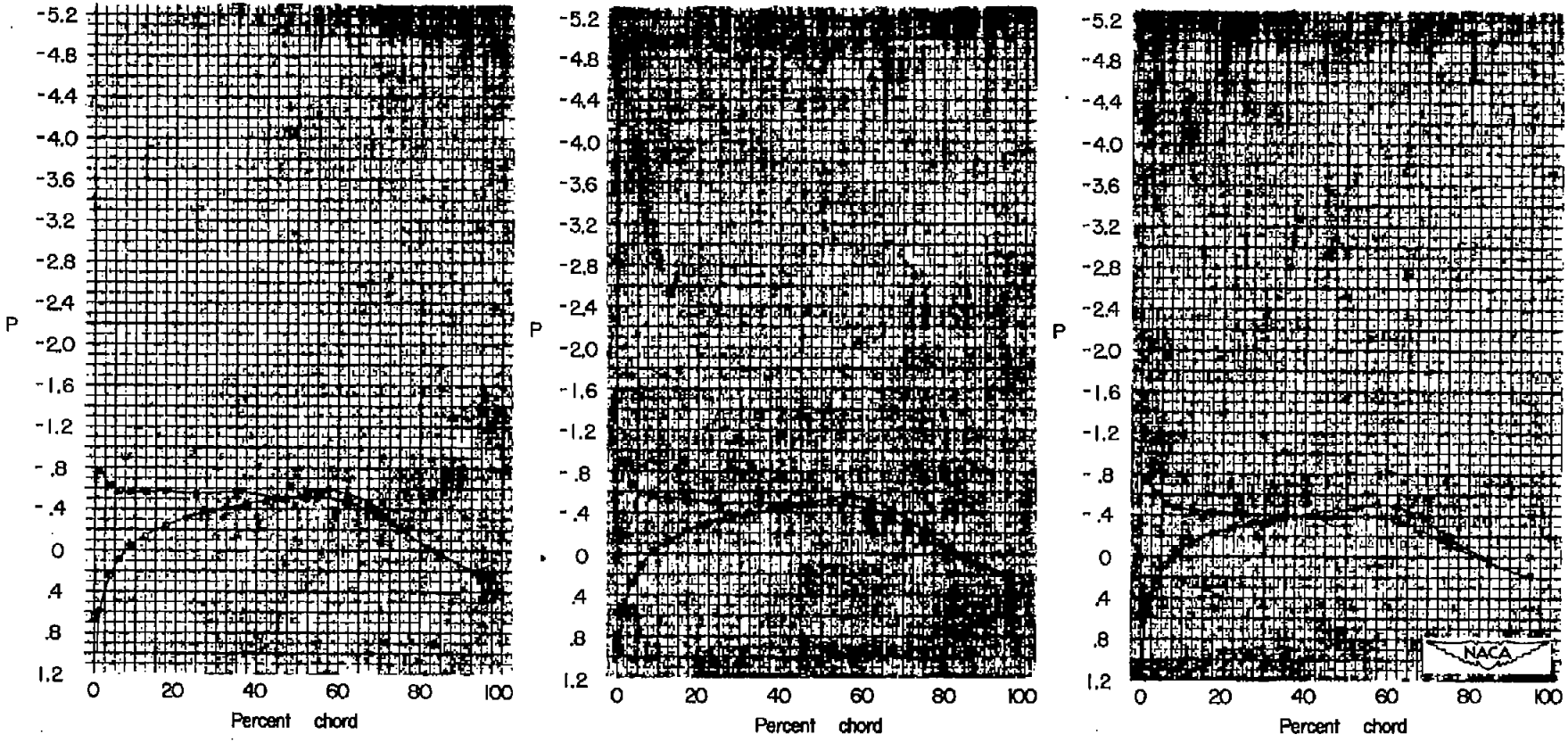
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(b) $\alpha = -1.95^{\circ}$.

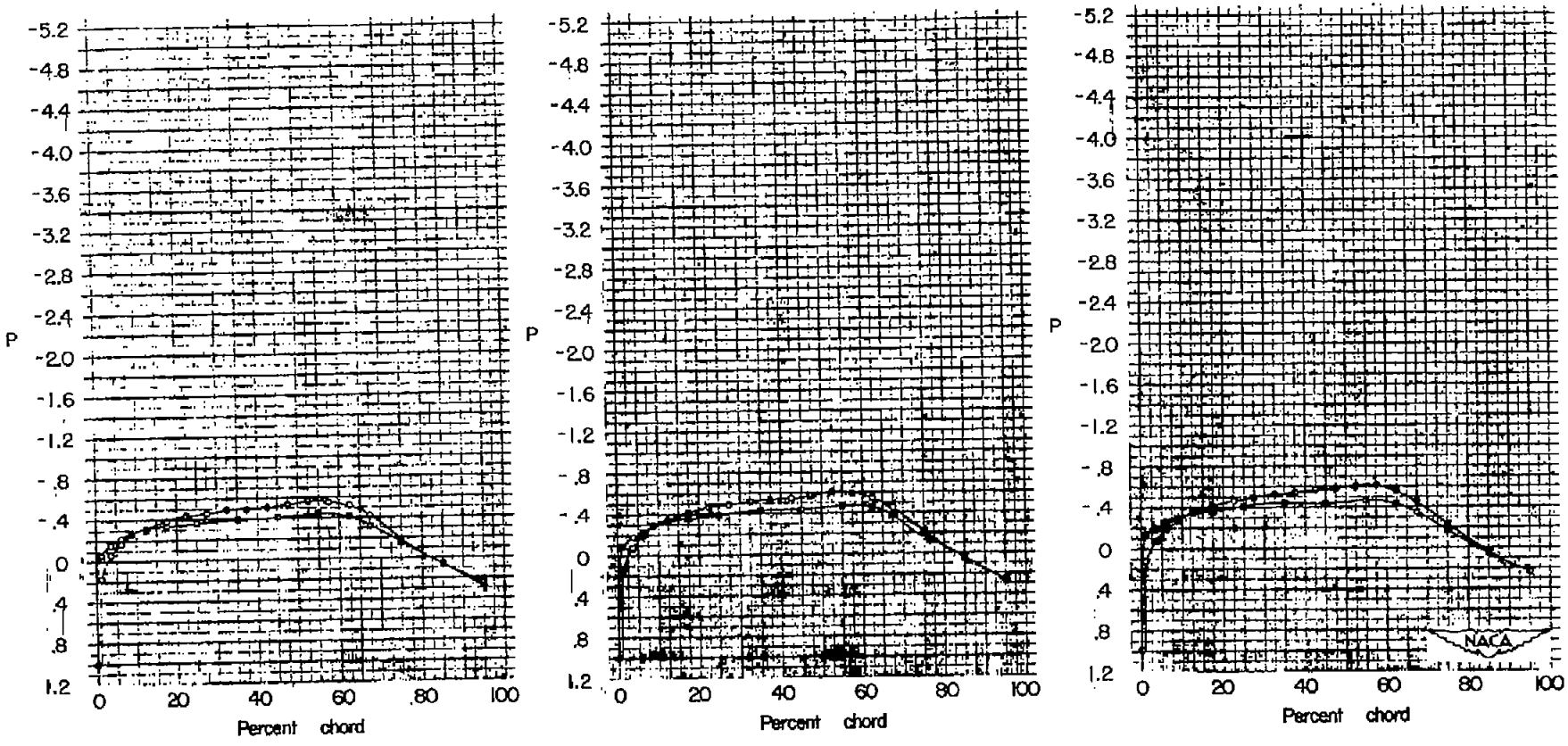
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = -1.95^{\circ}$.

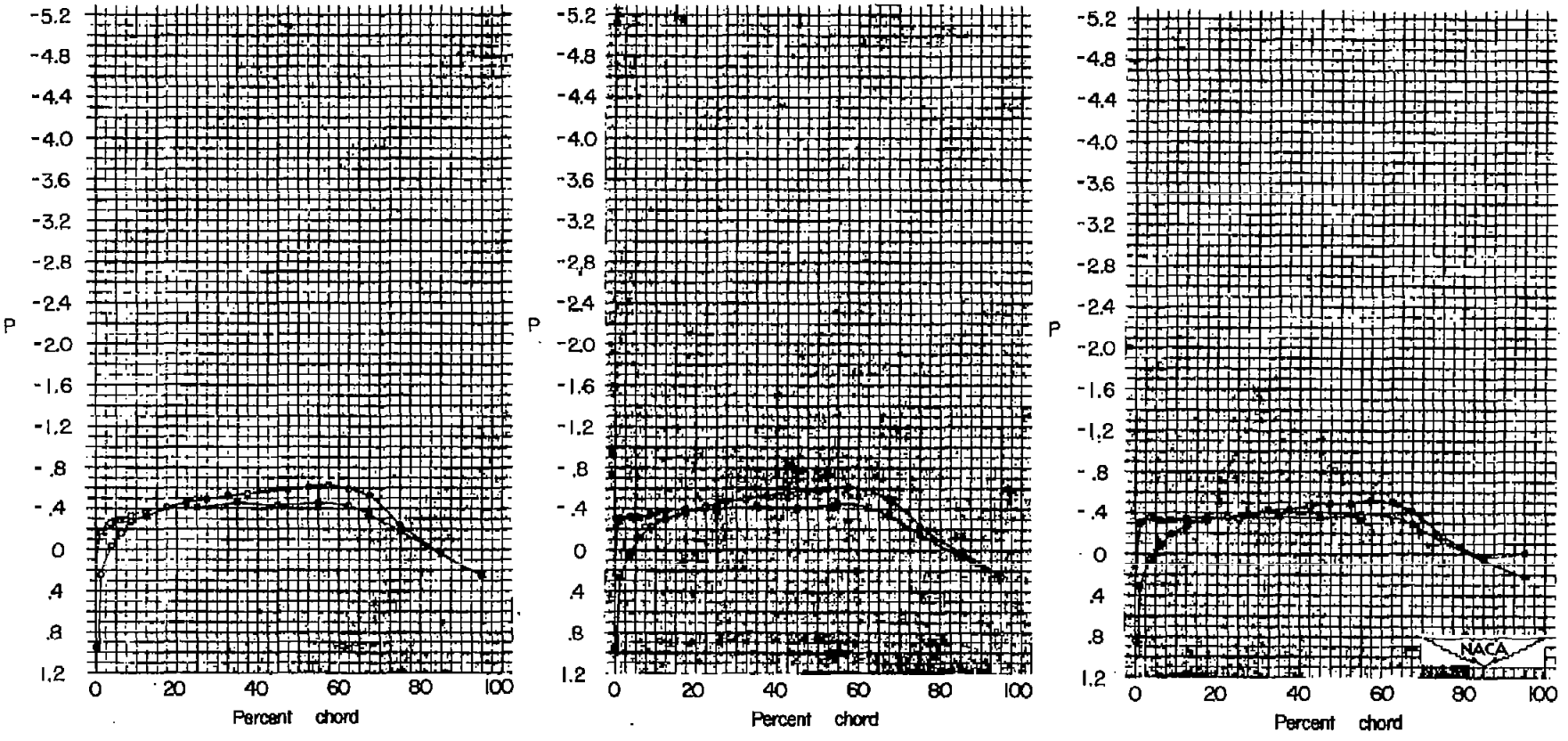
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(c) $\alpha = 0.26^{\circ}$.

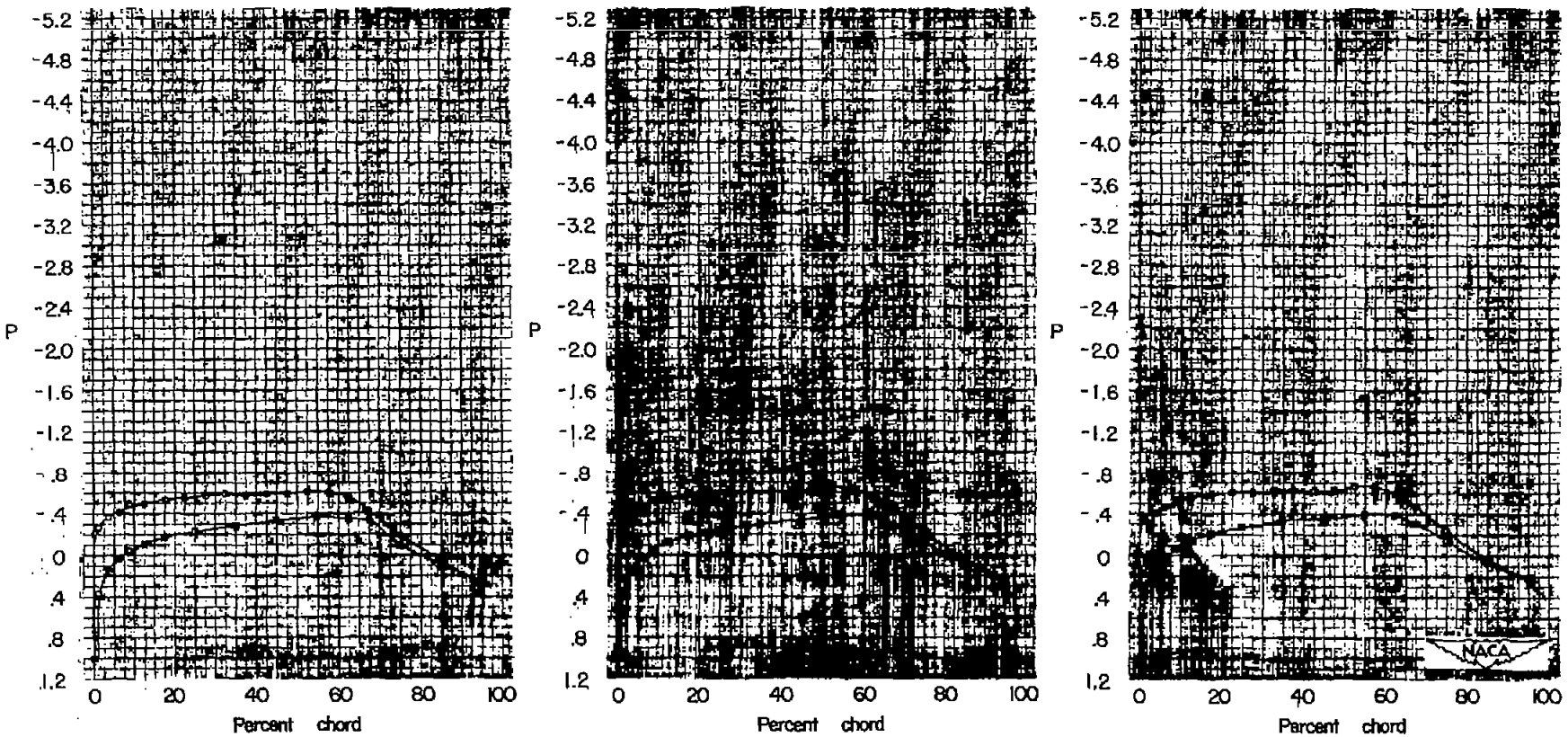
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 0.26^{\circ}$.

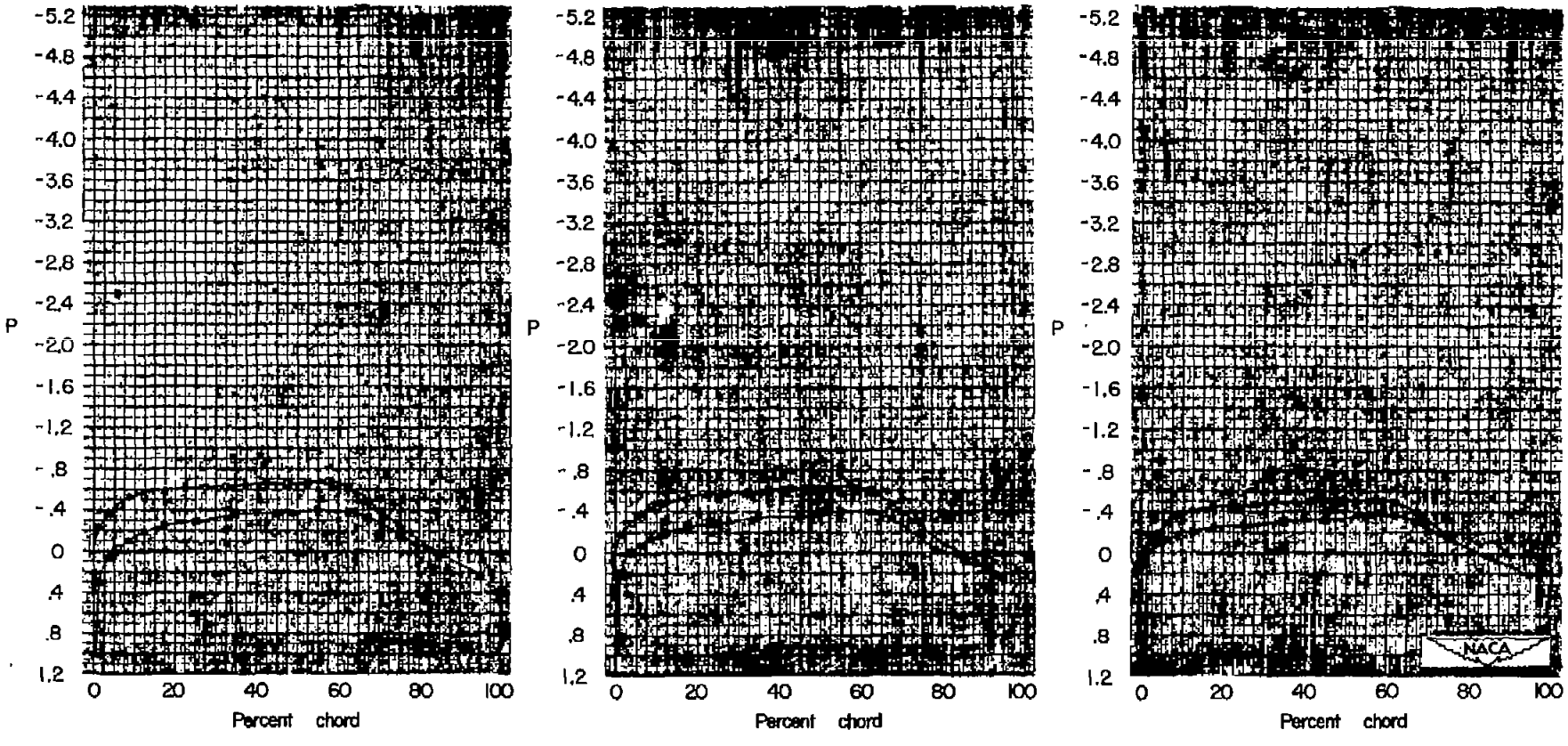
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(d) $\alpha = 2.42^{\circ}$.

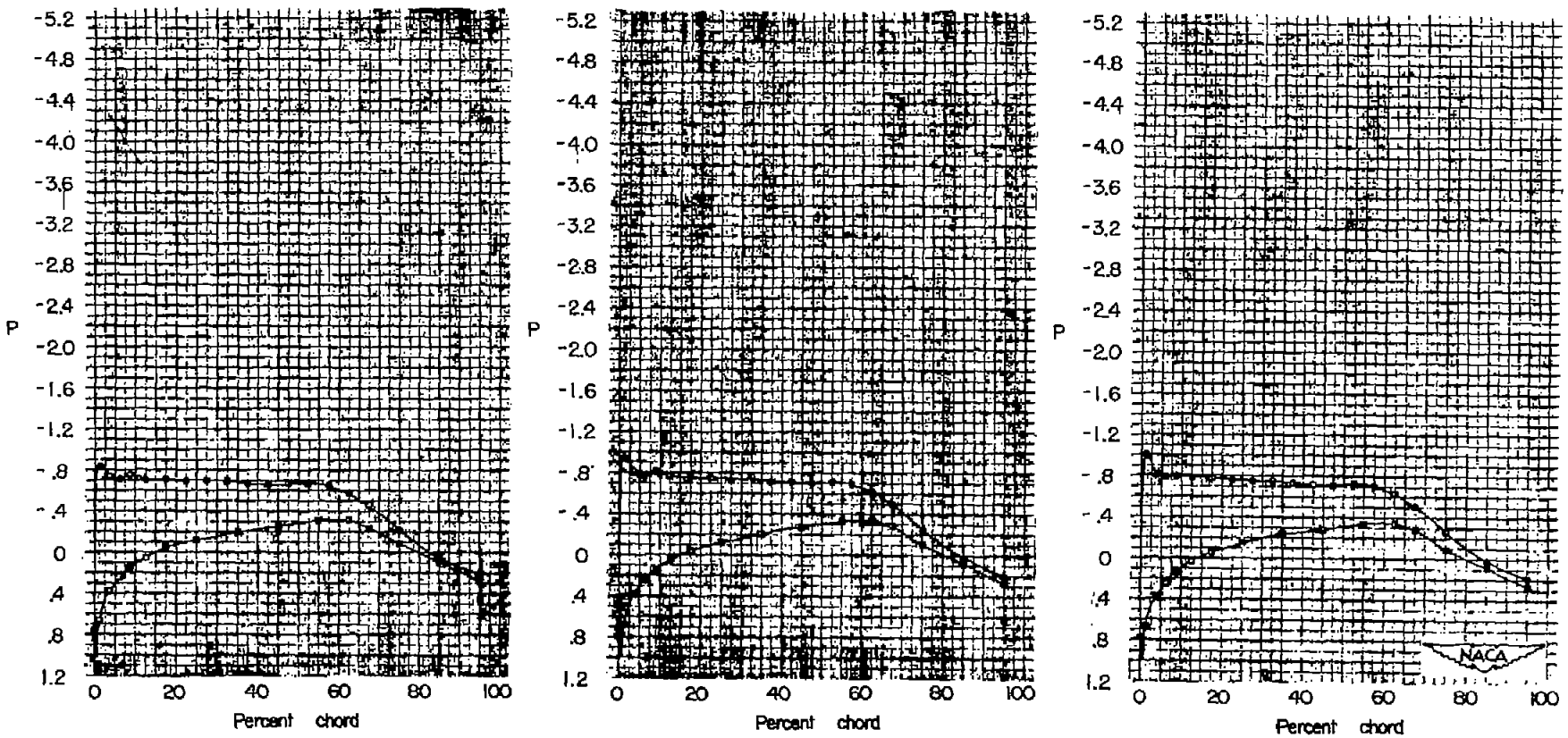
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(d) Concluded. $\alpha = 2.42^{\circ}$.

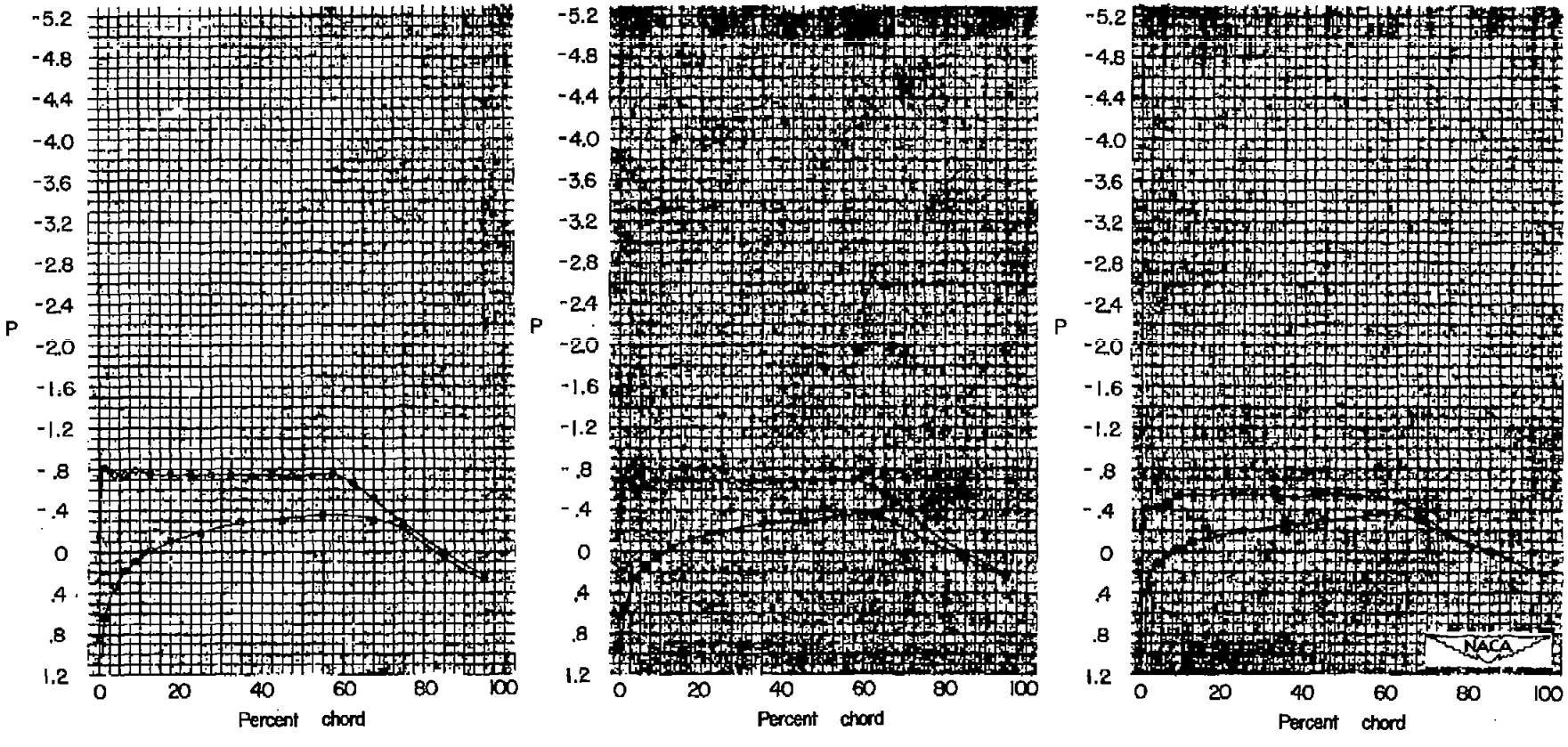
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(e) $\alpha = 4.58^{\circ}$.

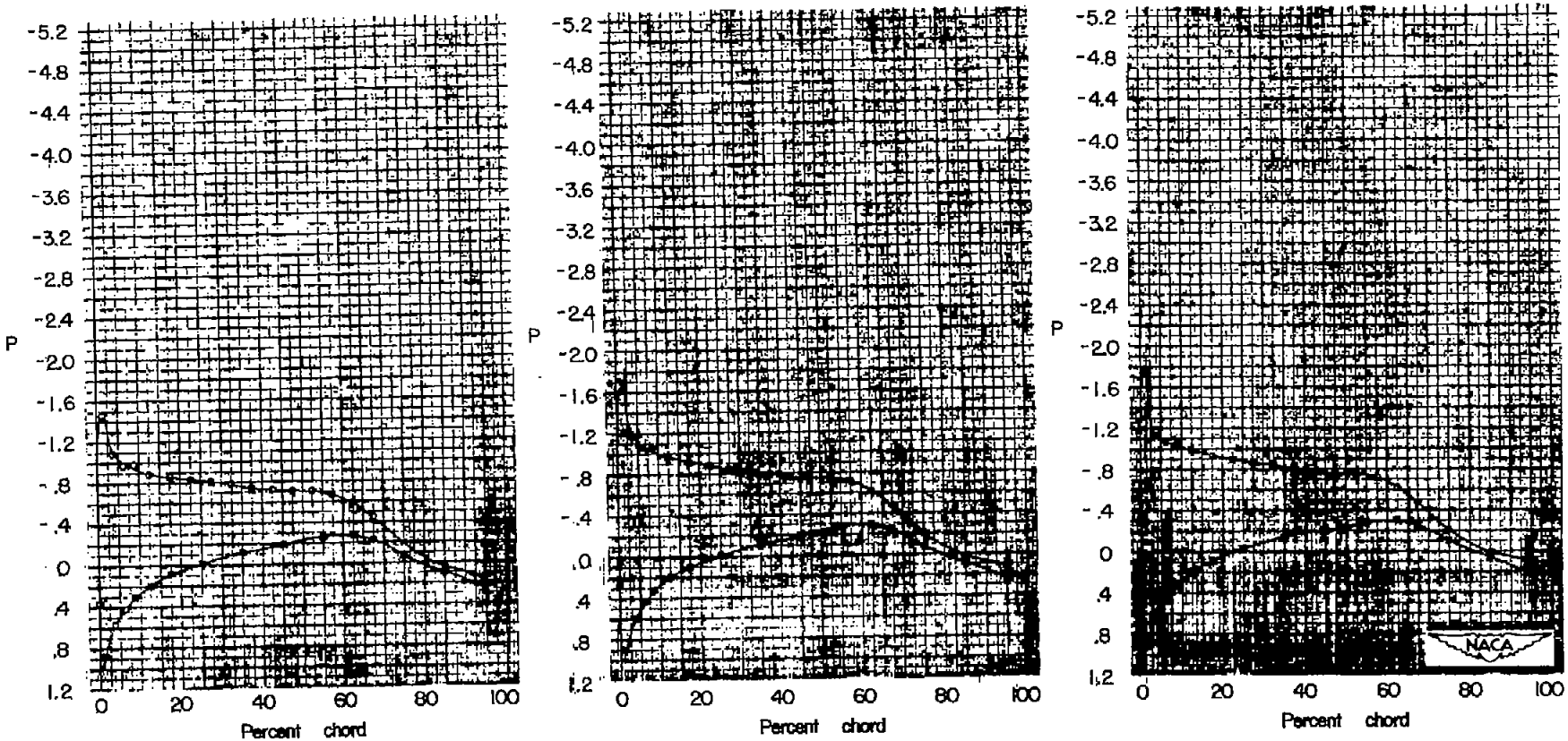
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 4.58^{\circ}$.

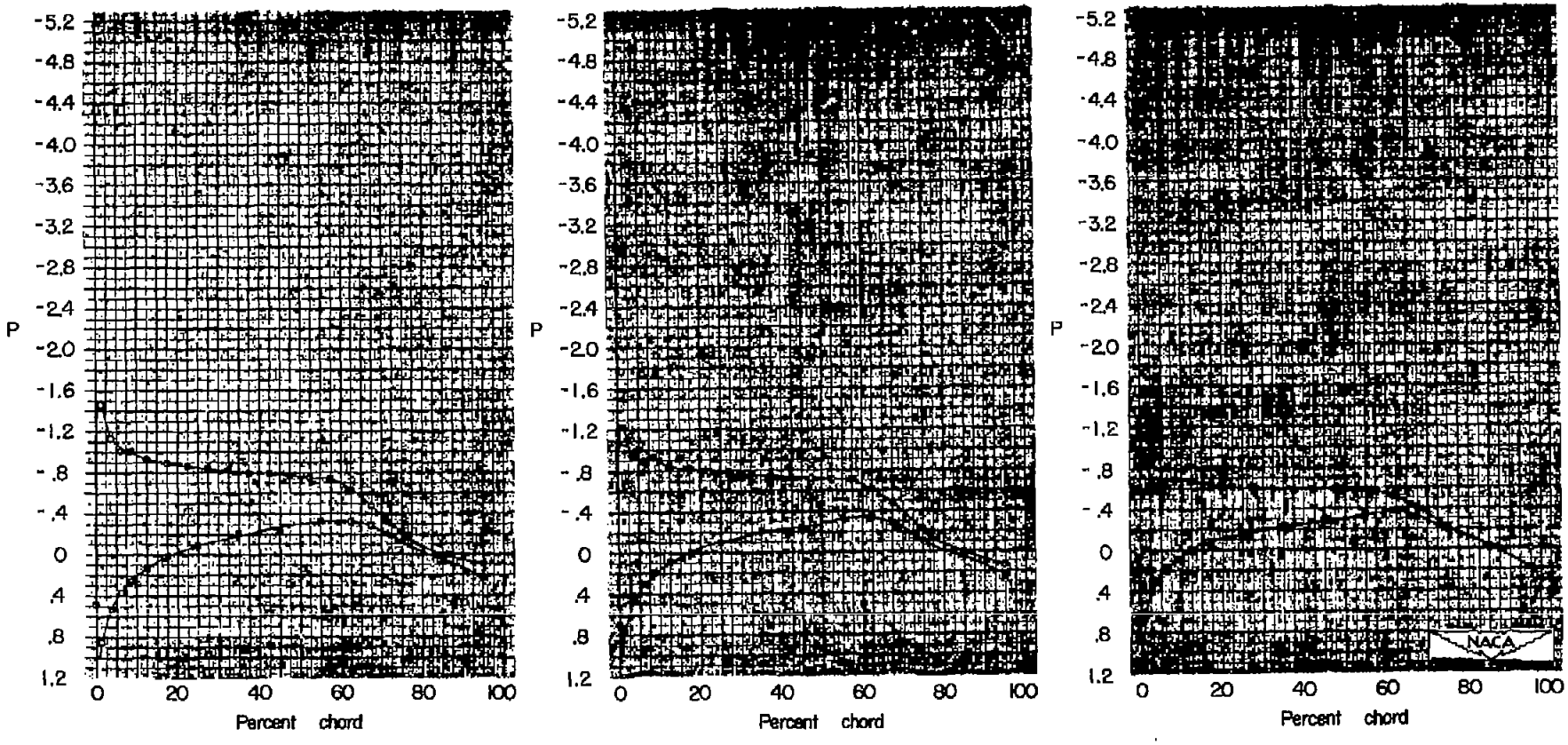
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(f) $\alpha = 6.74^{\circ}$.

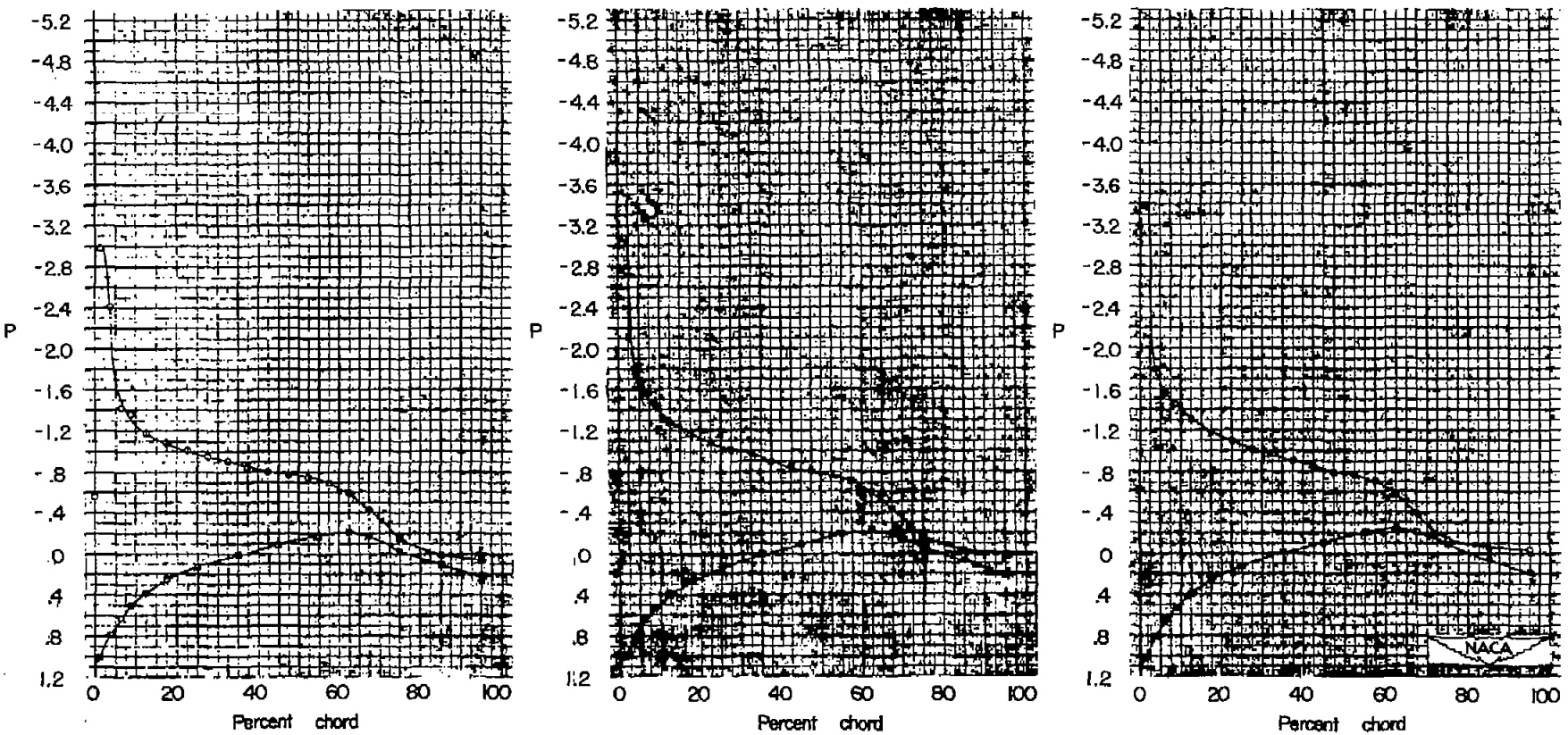
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(f) Concluded. $\alpha = 6.74^{\circ}$.

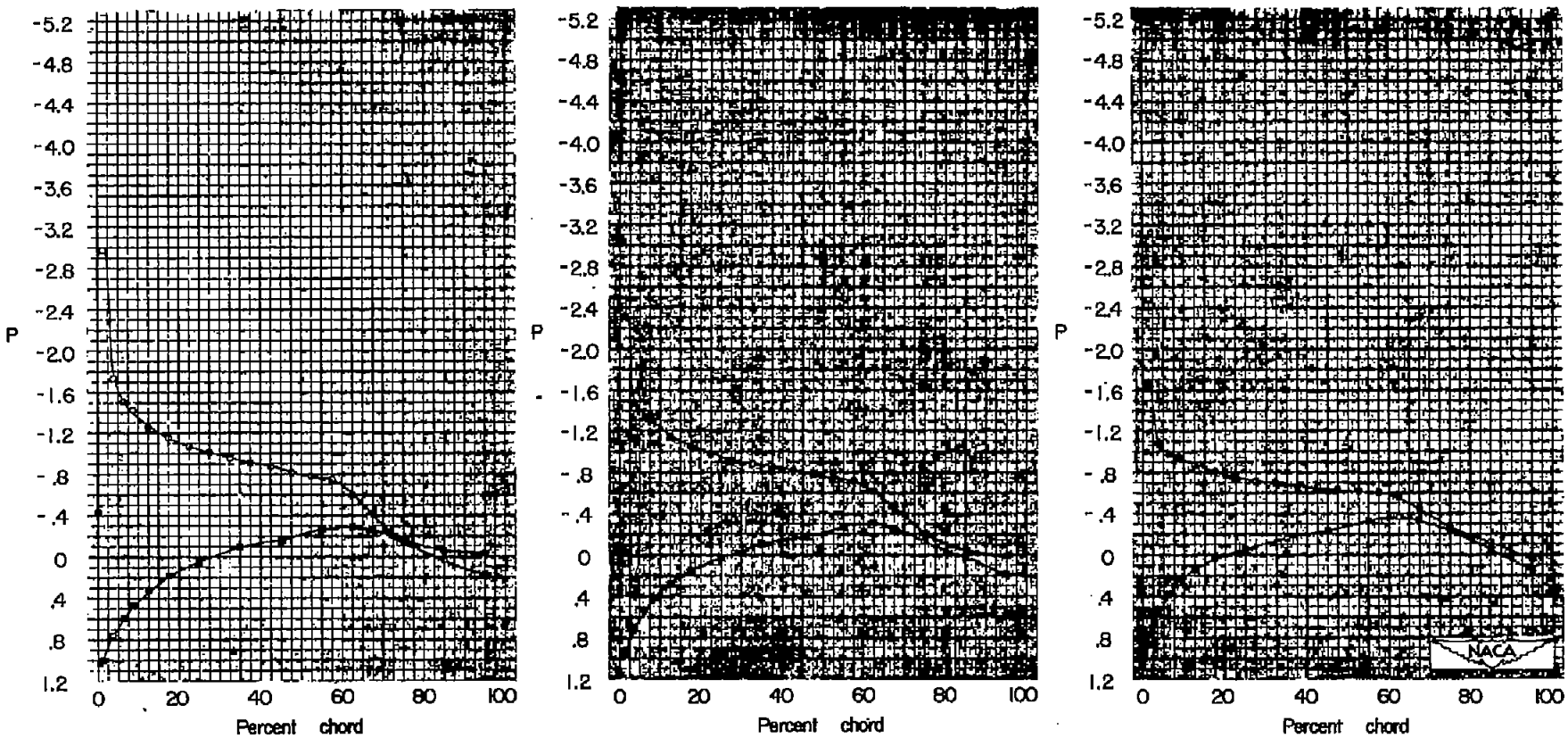
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(g) $\alpha = 9.98^{\circ}$.

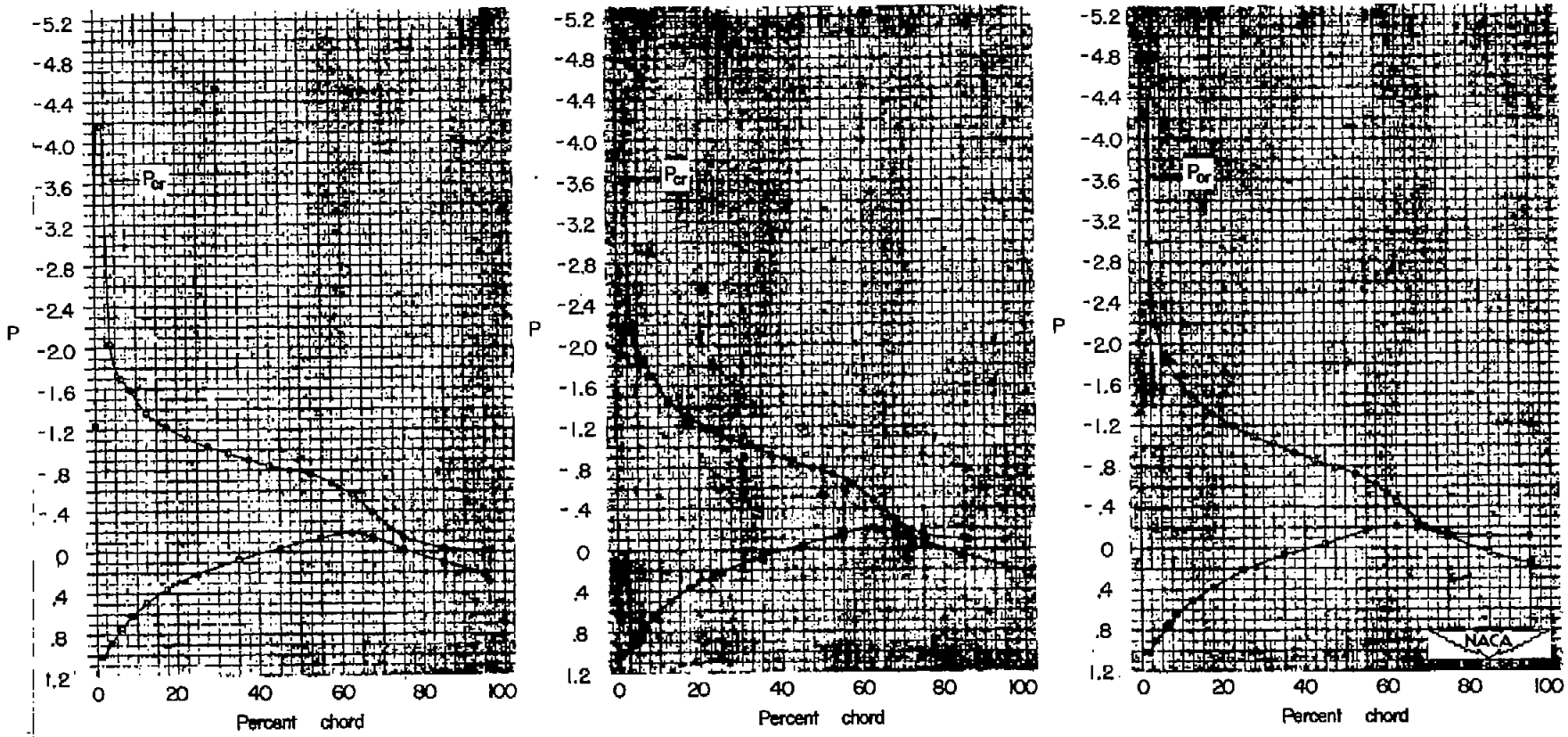
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(g) Concluded. $\alpha = 9.98^{\circ}$.

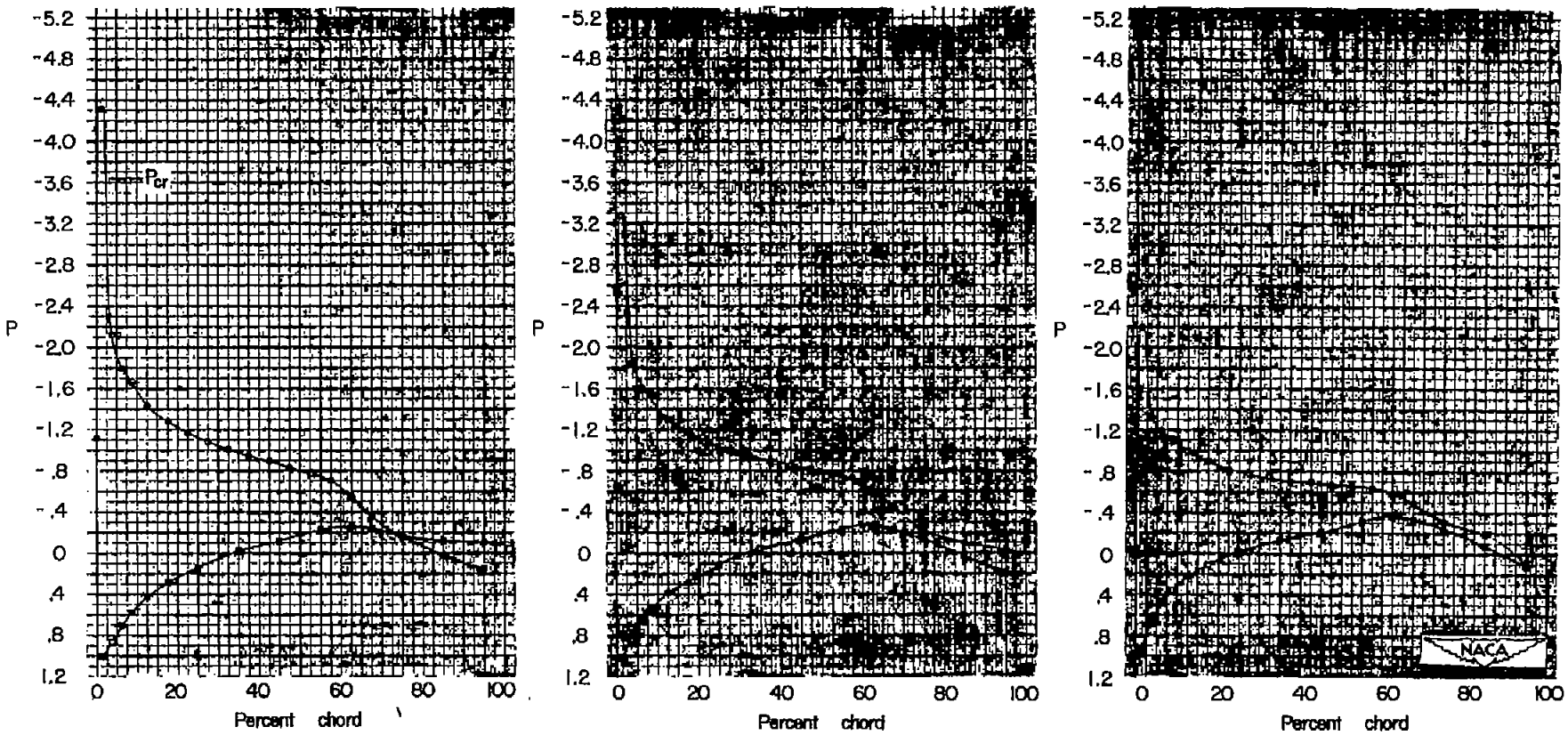
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(h) $\alpha = 12.10^{\circ}$.

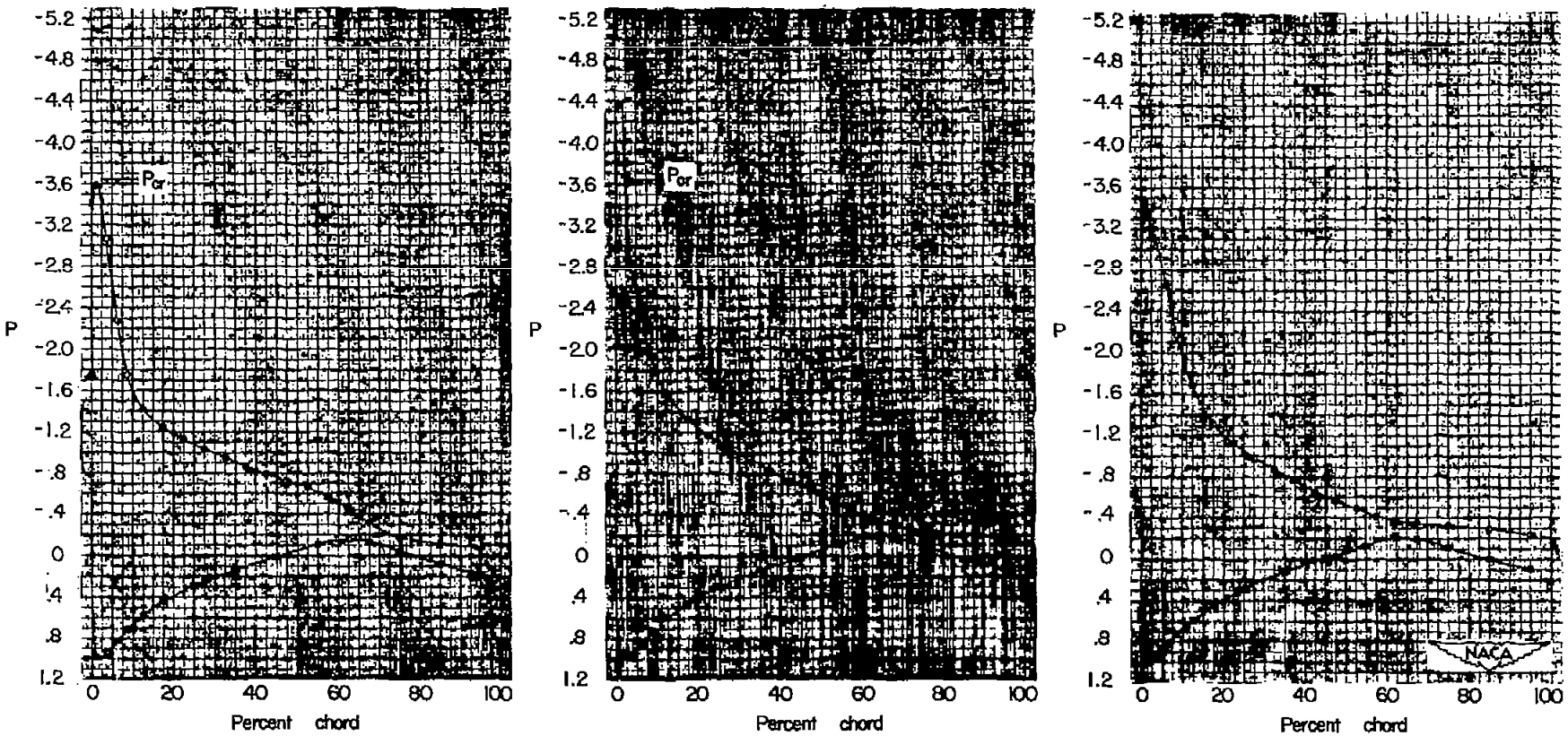
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(h) Concluded. $\alpha = 12.10^{\circ}$.

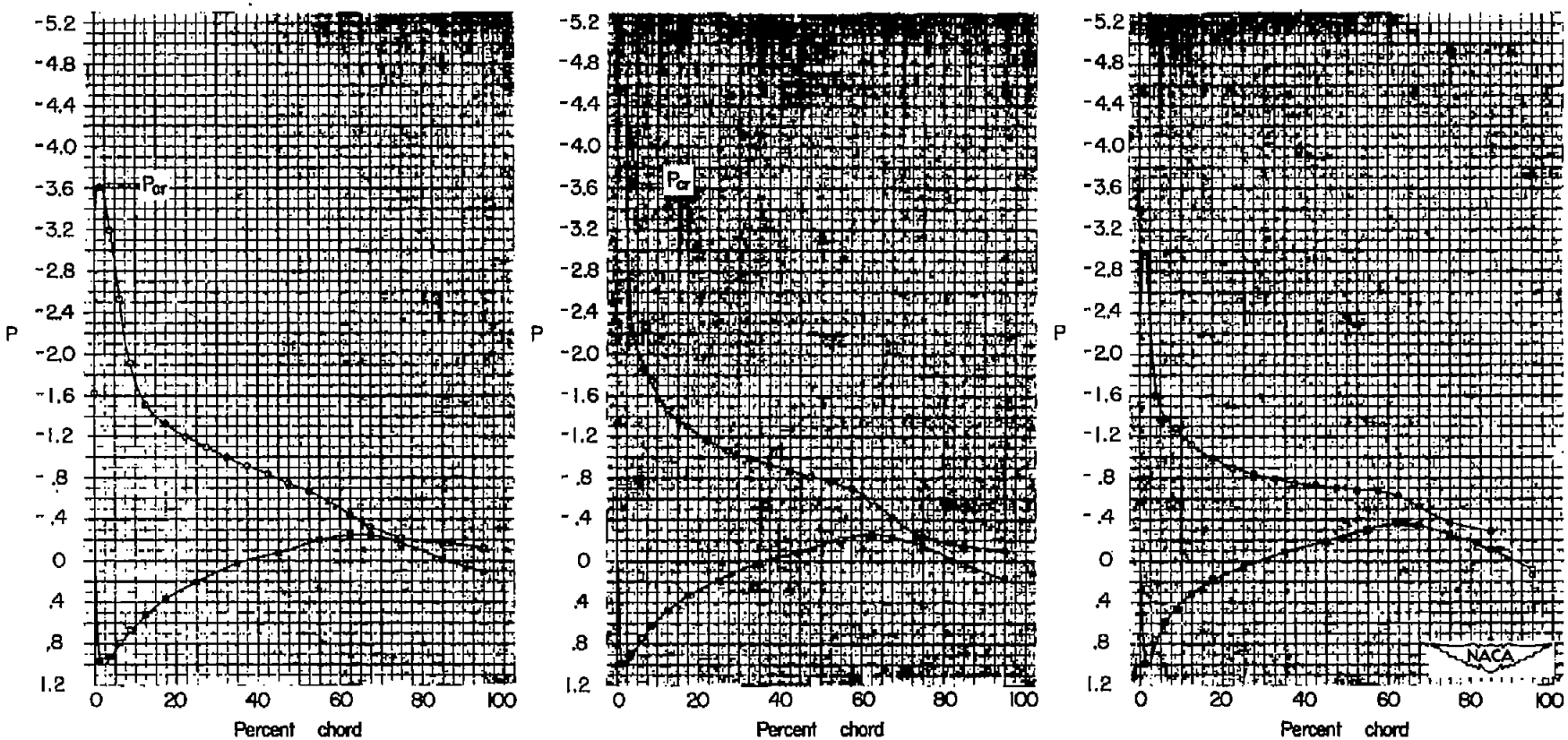
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(i) $\alpha = 14.59^{\circ}$.

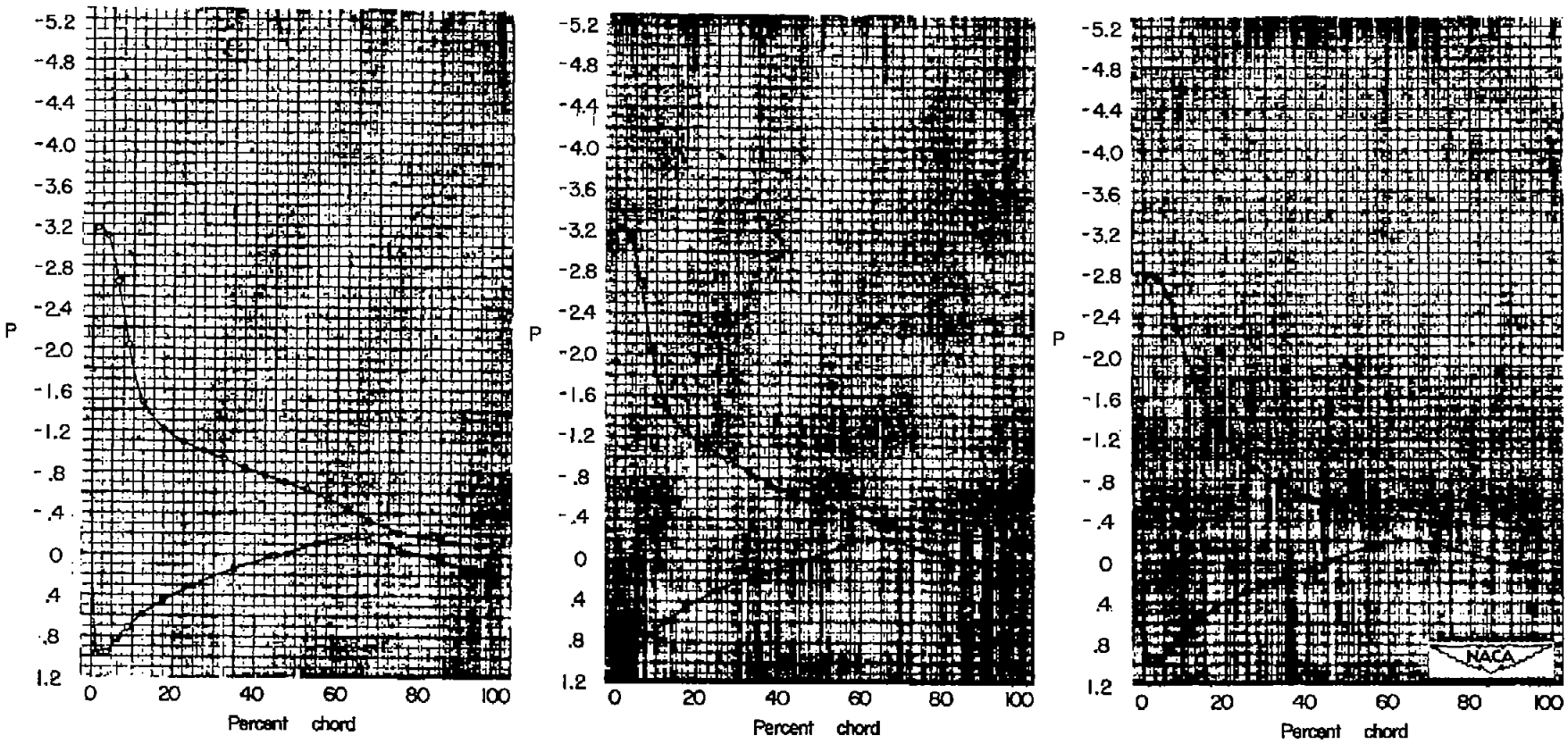
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(i) Concluded. $\alpha = 14.59^{\circ}$.

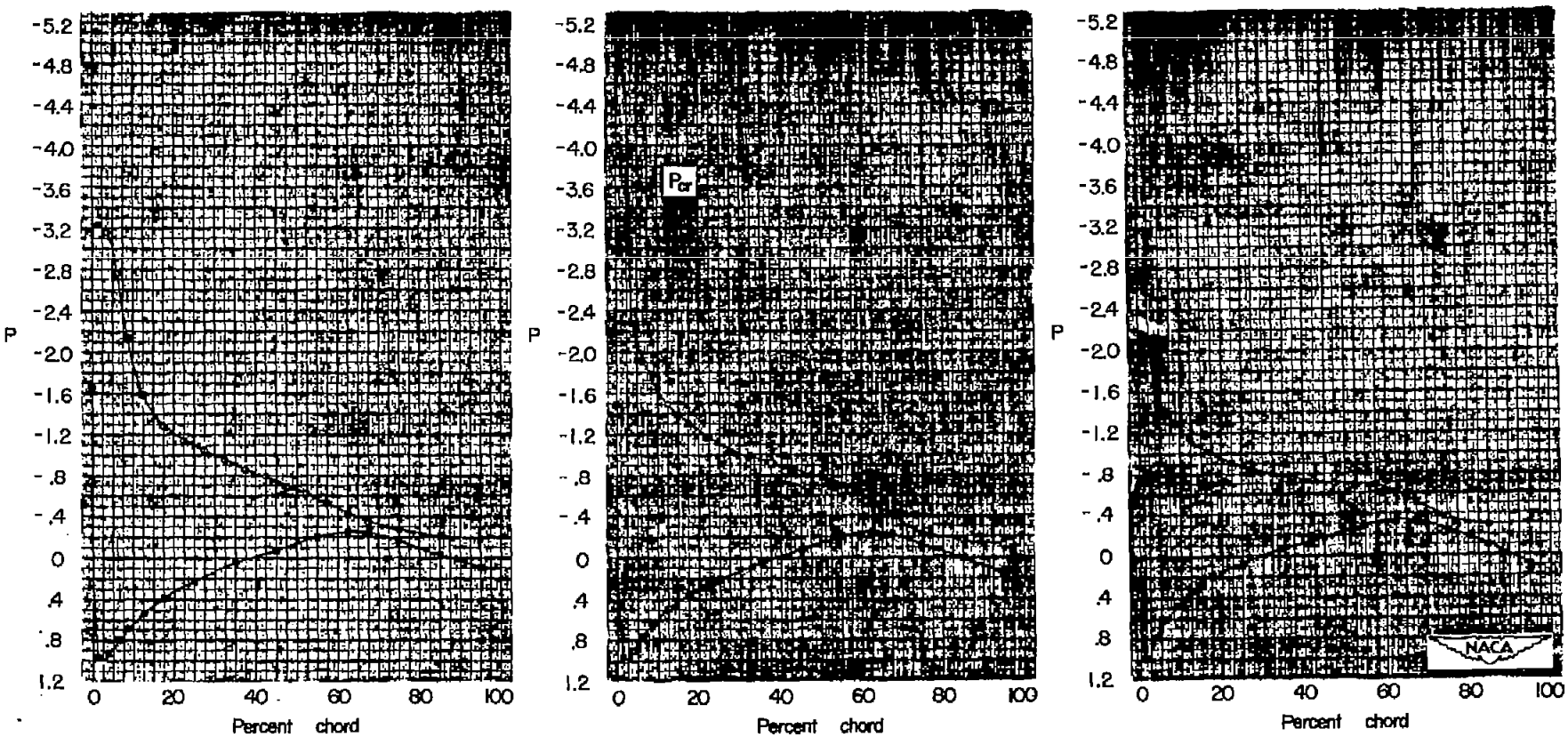
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(j) $\alpha = 15.22^{\circ}$.

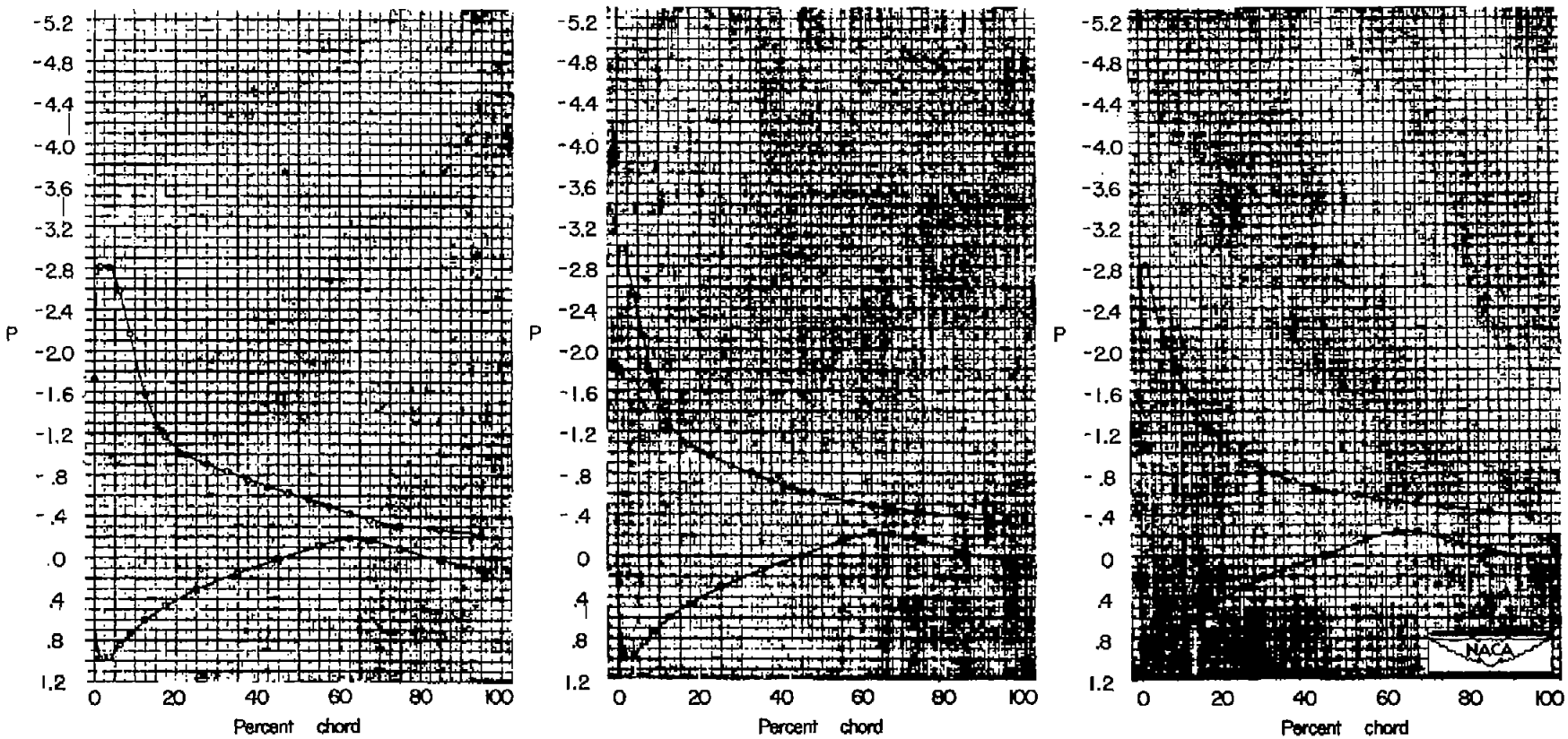
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(j) Concluded. $\alpha = 15.22^{\circ}$.

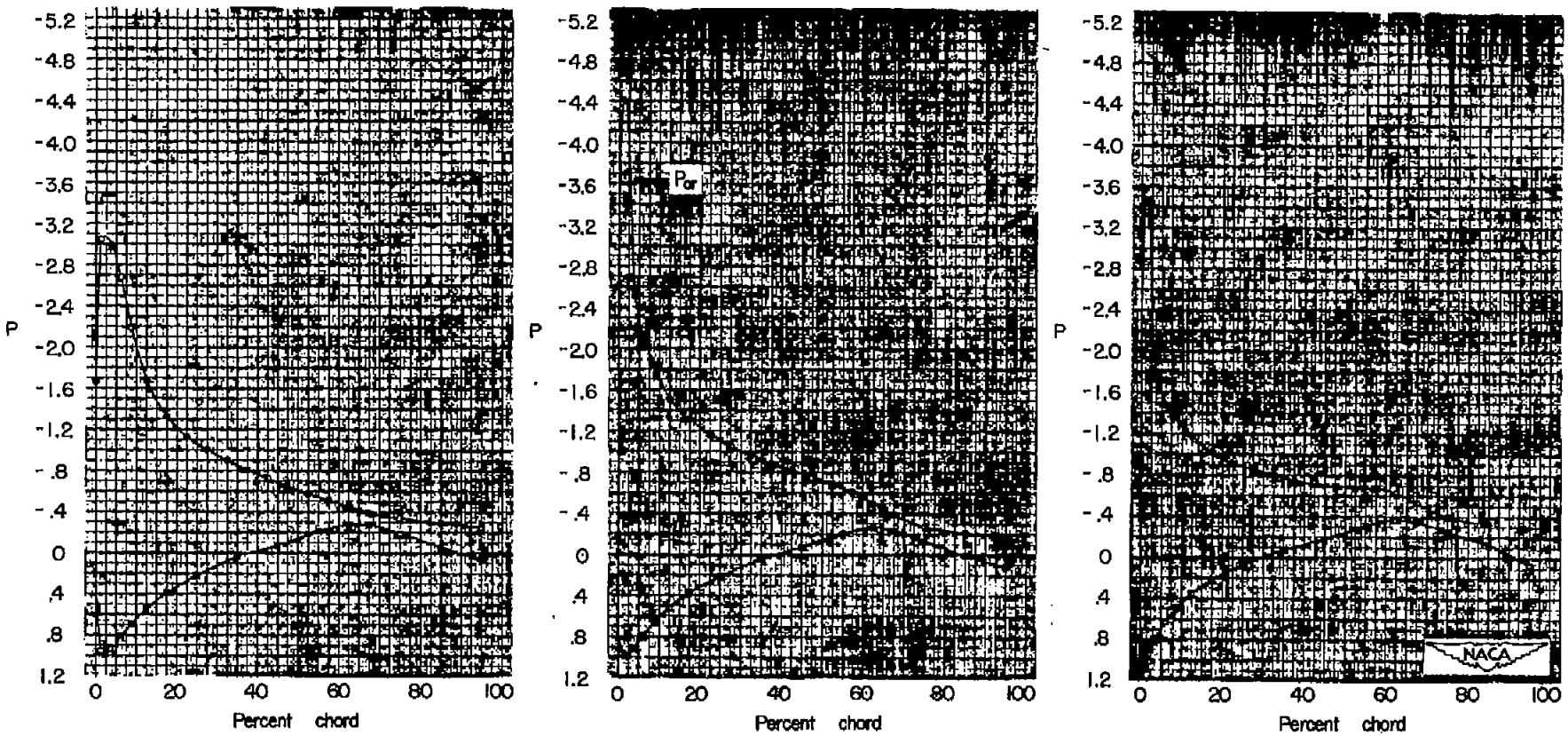
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(k) $\alpha = 16.17^{\circ}$.

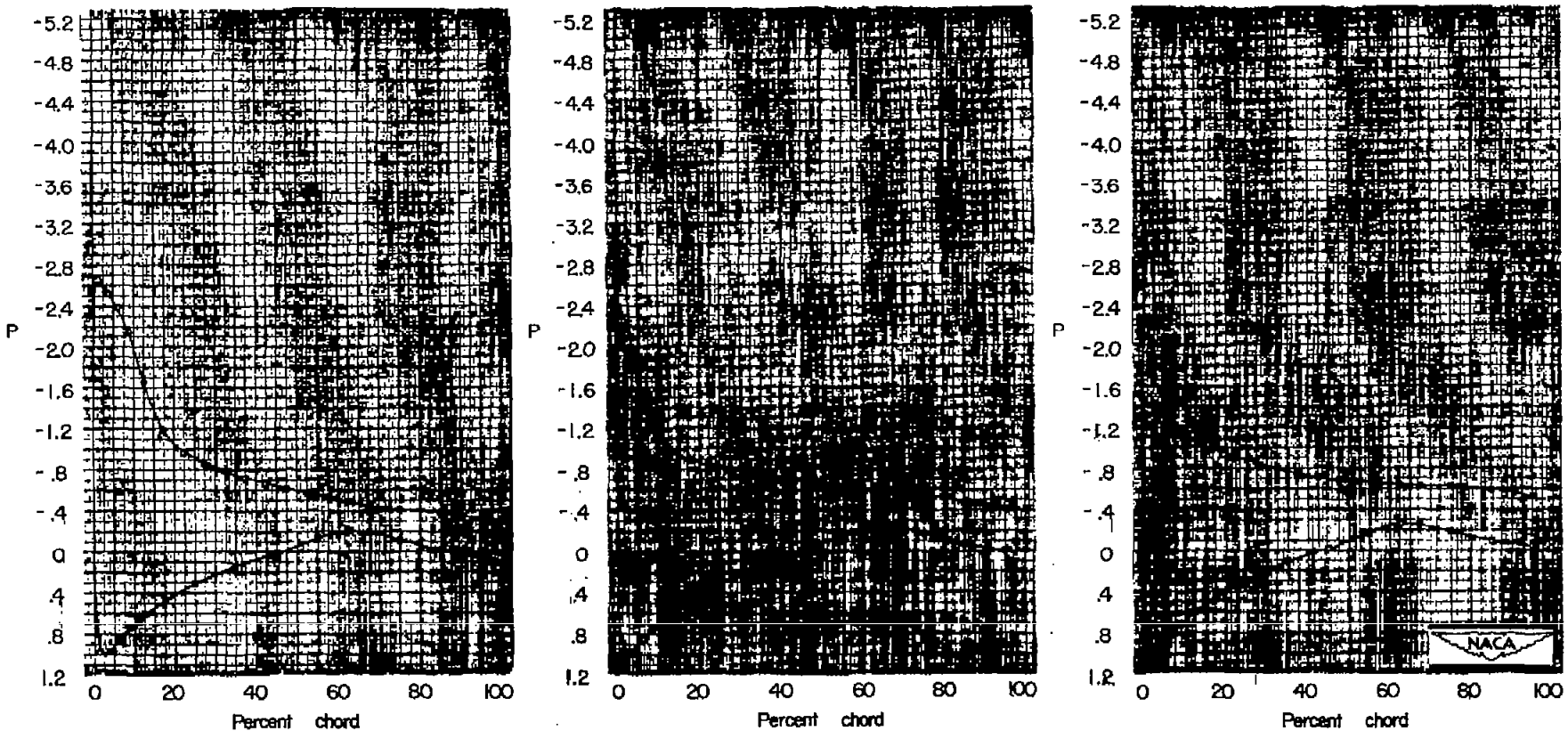
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(k) Concluded. $\alpha = 16.17^{\circ}$.

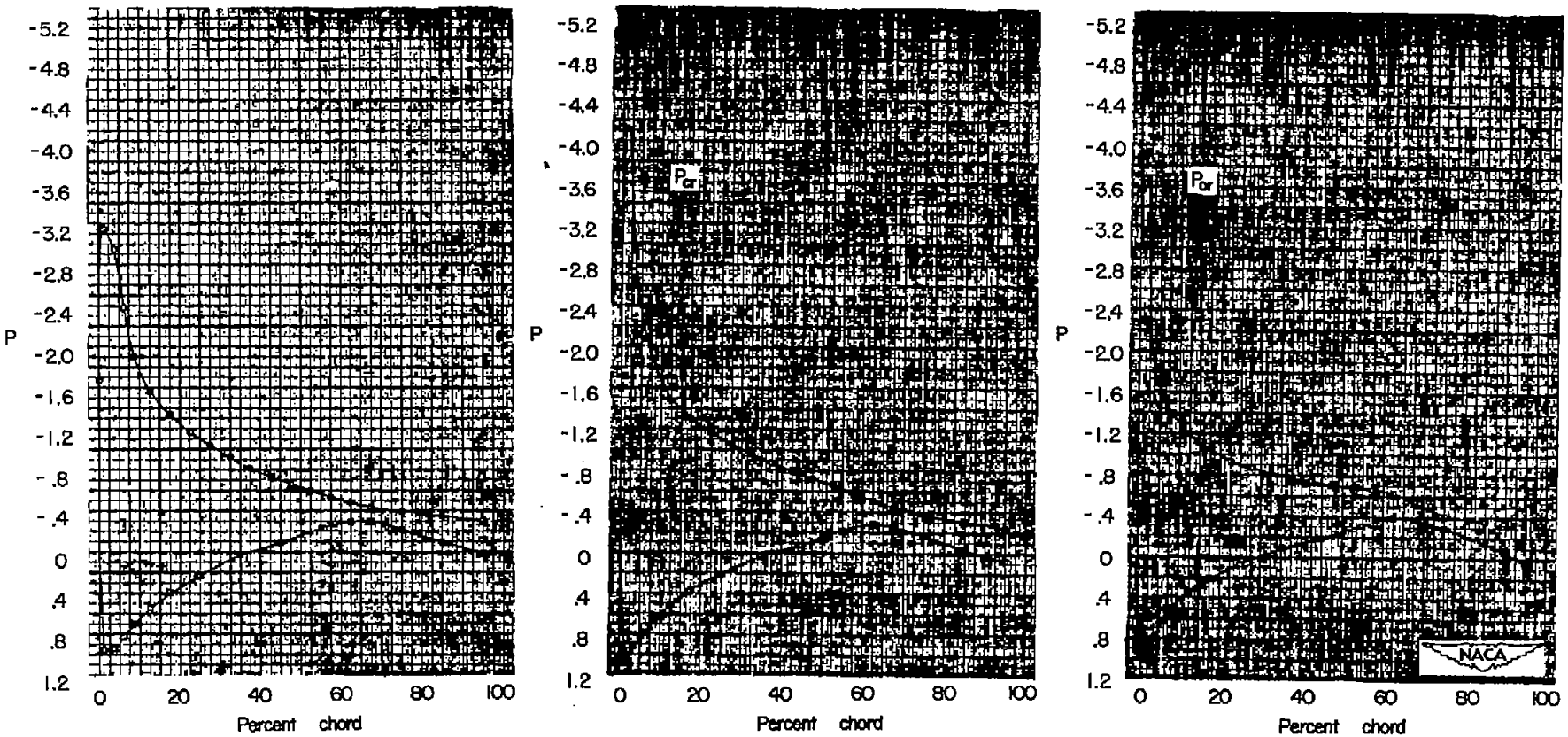
Figure 8.- Continued. M = 0.40.



Station 1 Station 2 Station 3

(l) $\alpha = 17.15^{\circ}$.

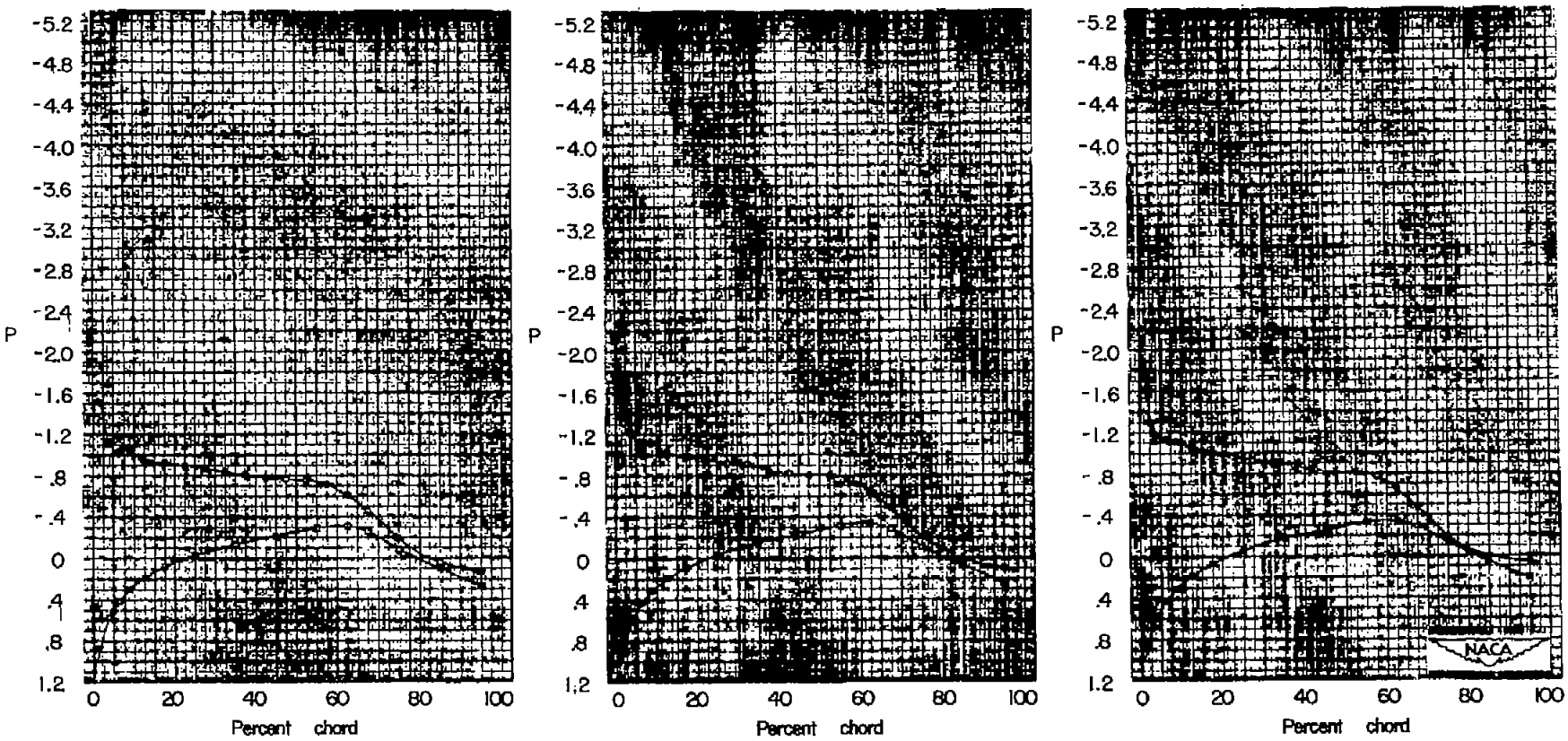
Figure 8.- Continued. M = 0.40.



Station 4 Station 5 Station 6

(*l*) Concluded. $\alpha = 17.15^{\circ}$.

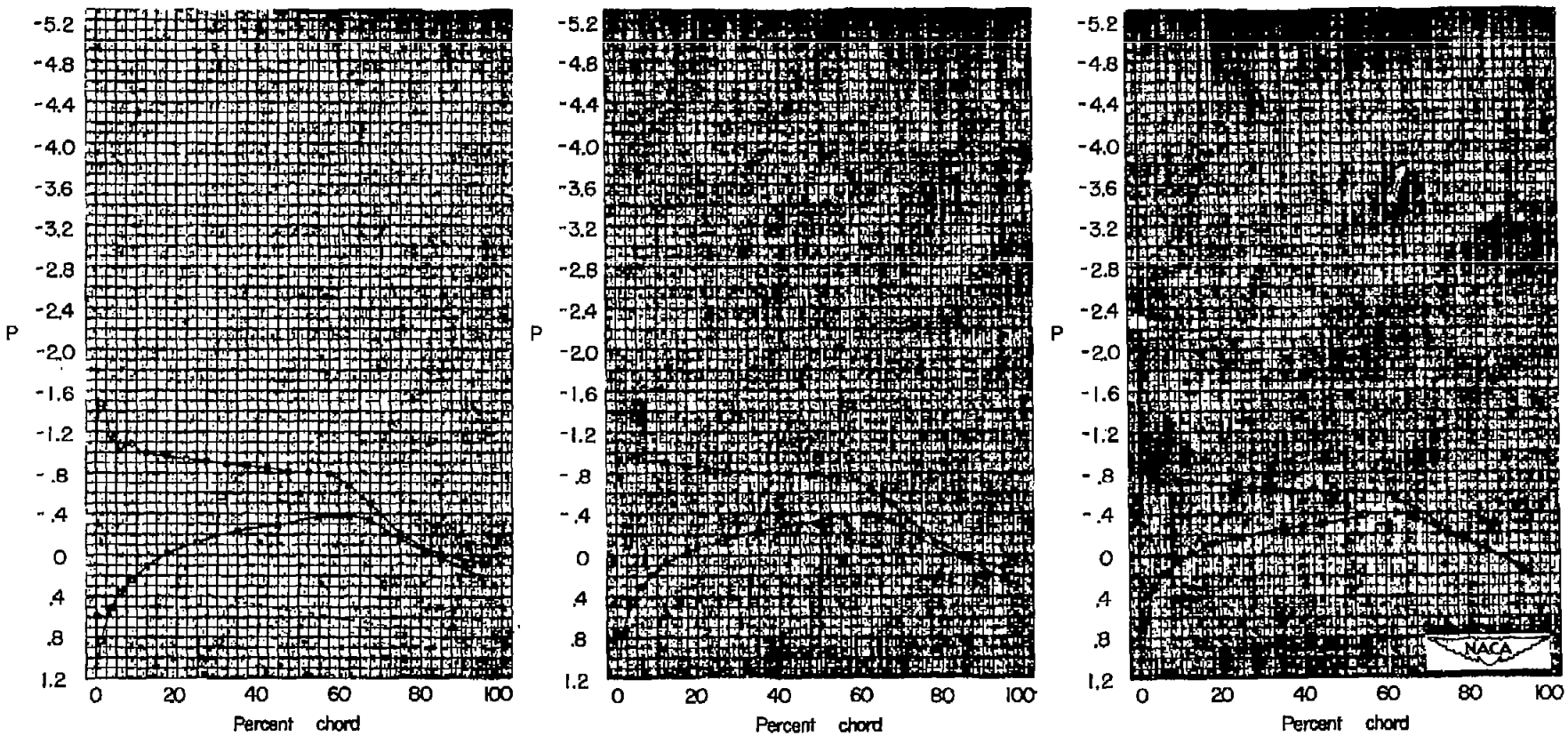
Figure 8.- Concluded. M = 0.40.



Station 1 Station 2 Station 3

(a) $\alpha = 6.75^{\circ}$.

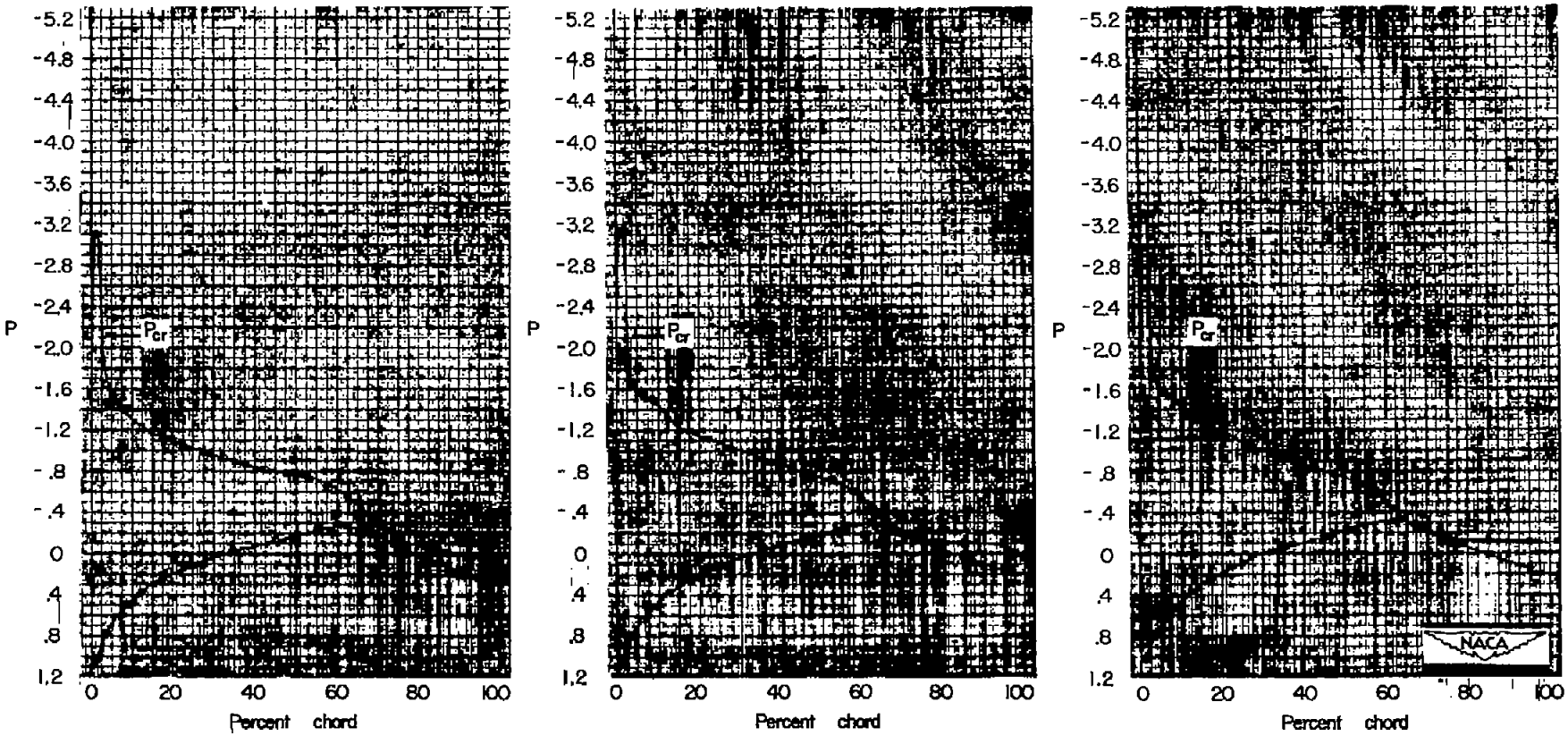
Figure 9.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. $M = 0.50$.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = 6.75^{\circ}$.

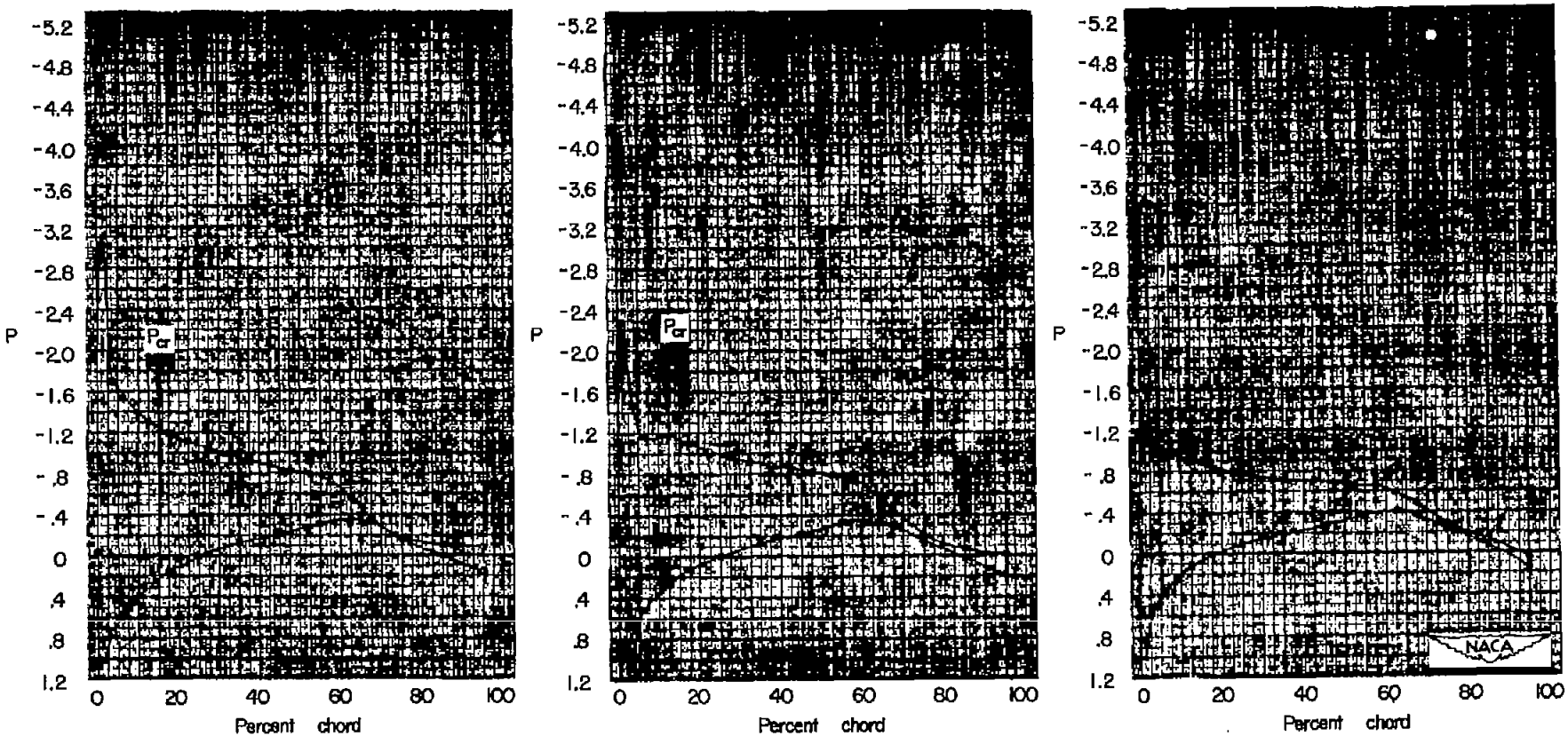
Figure 9.- Continued. M = 0.50.



Station 1 Station 2 Station 3

(b) $\alpha = 9.98^{\circ}$.

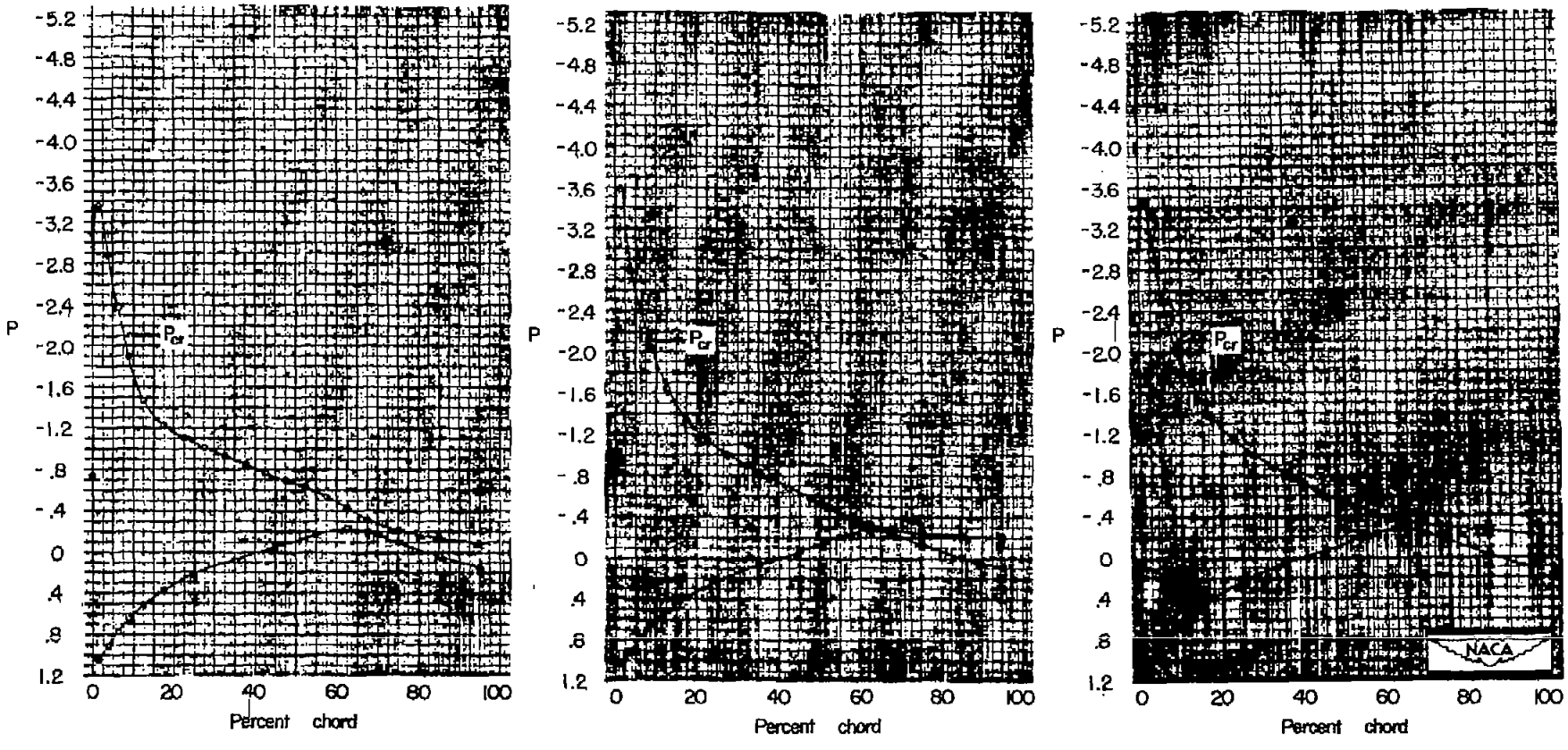
Figure 9.- Continued. M = 0.50.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = 9.98^{\circ}$.

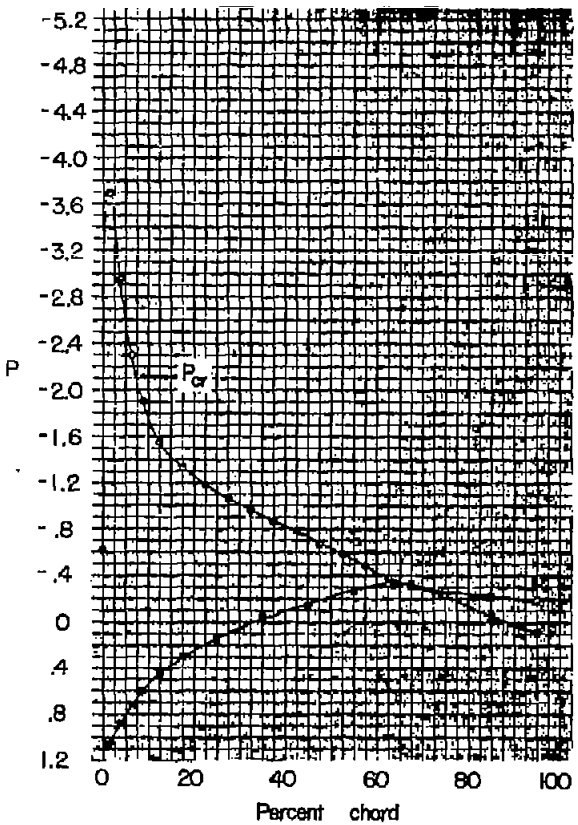
Figure 9.- Continued. M = 0.50.



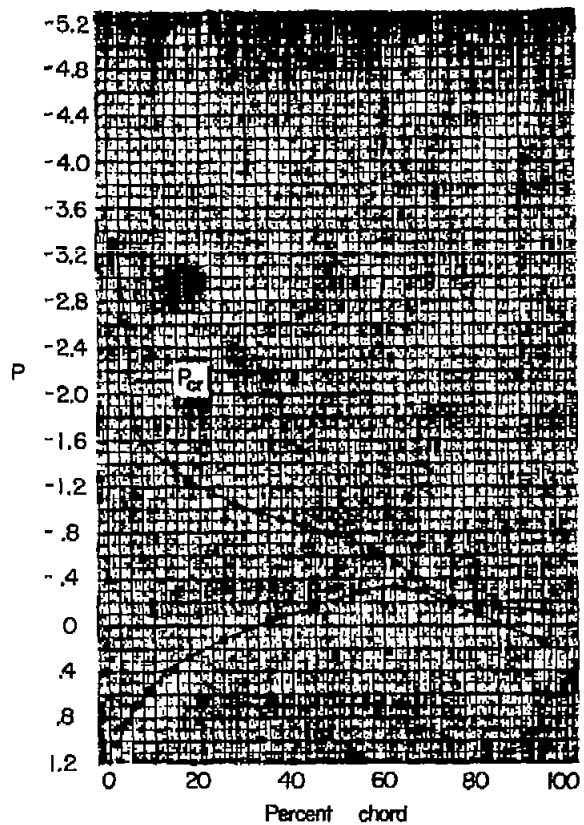
Station 1 Station 2 Station 3

(c) $\alpha = 13.54^{\circ}$.

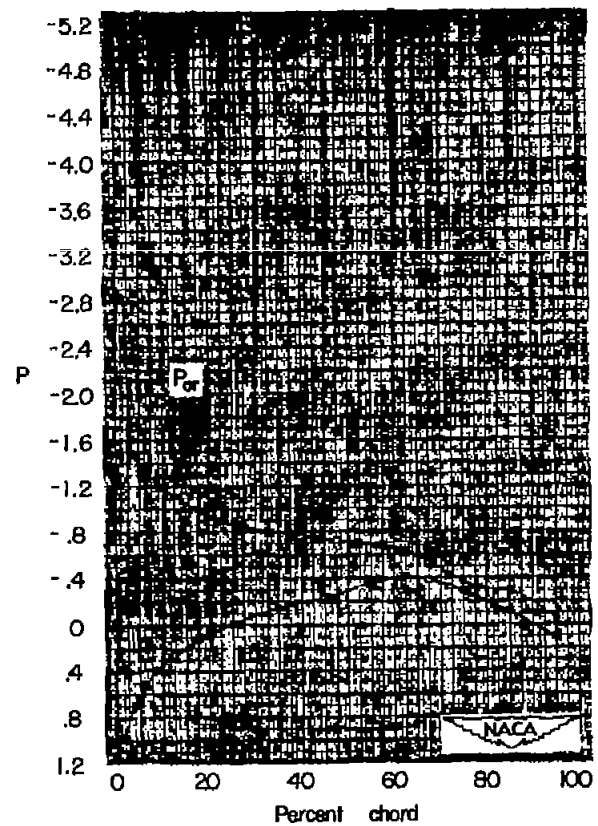
Figure 9.- Continued. M = 0.50.



Station 4



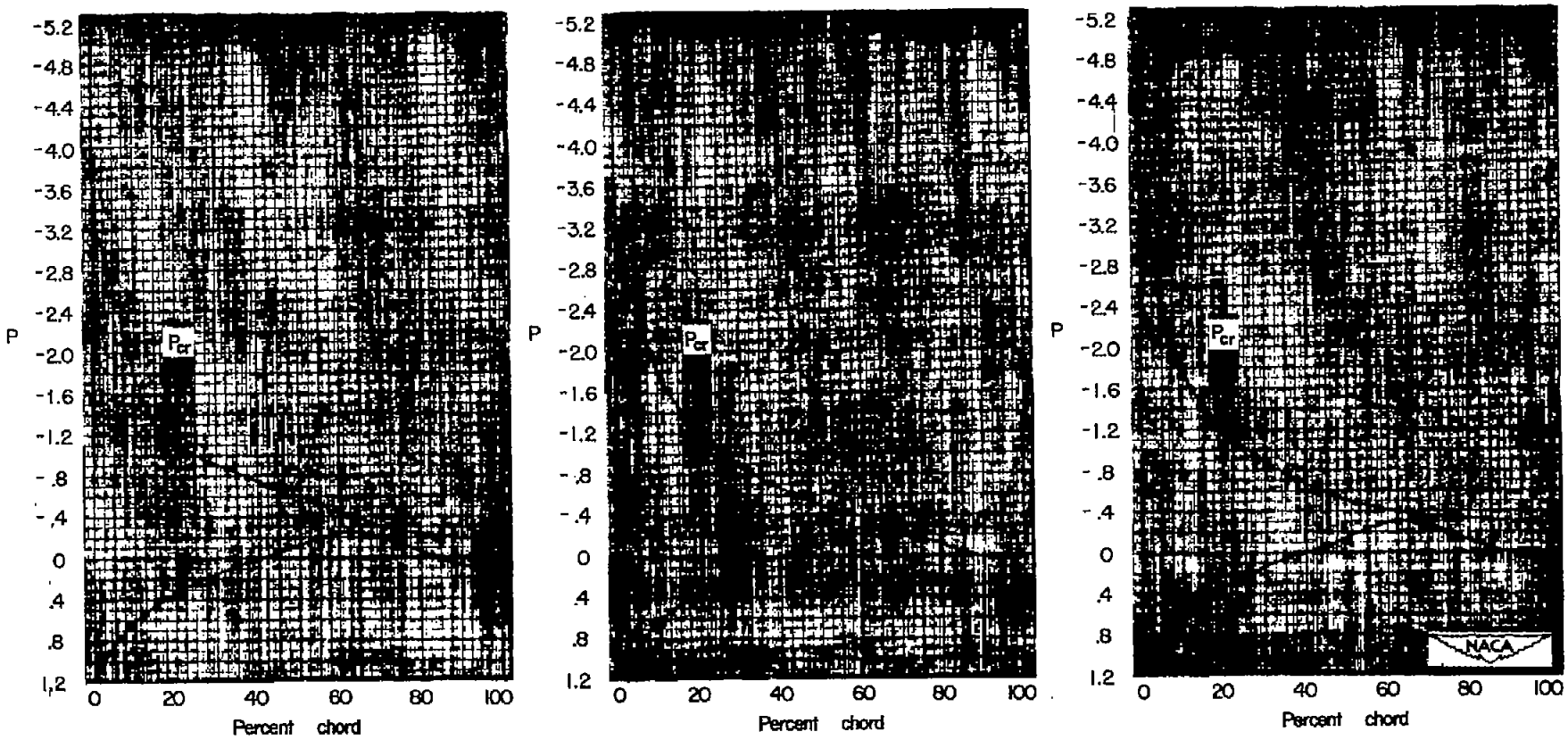
Station 5



Station 6

(c) Concluded. $\alpha = 13.54^{\circ}$.

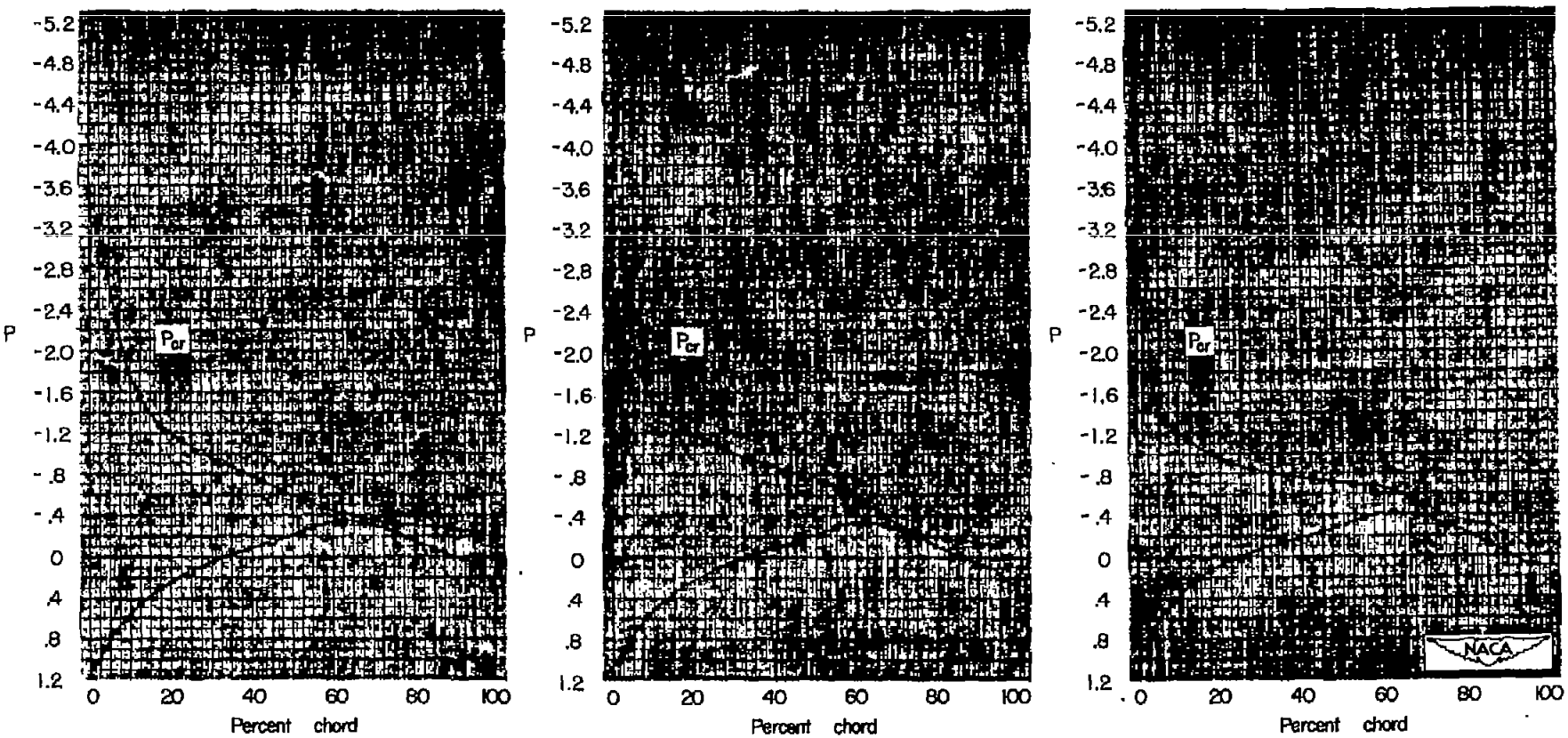
Figure 9.- Continued. $M = 0.50$.



Station 1 Station 2 Station 3

(d) $\alpha = 14.17^{\circ}$.

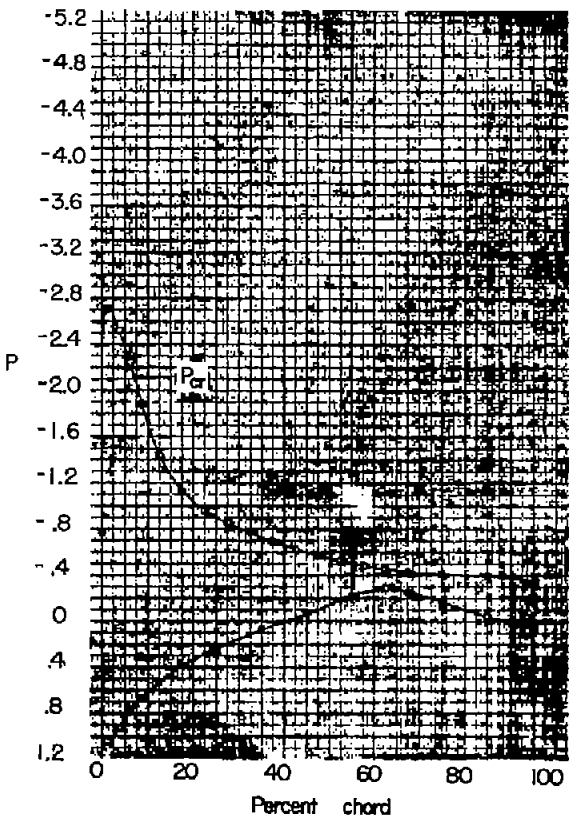
Figure 9.- Continued. M = 0.50.



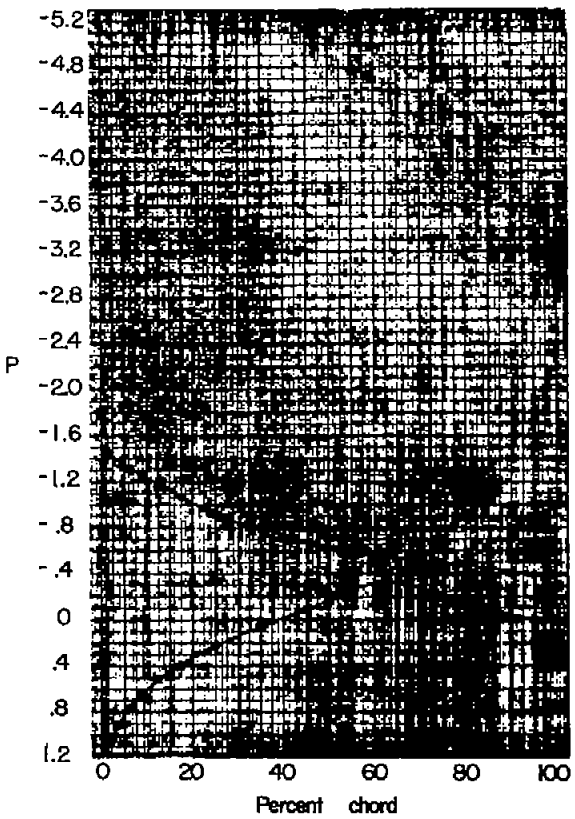
Station 4 Station 5 Station 6

(d) Concluded. $\alpha = 14.17^{\circ}$.

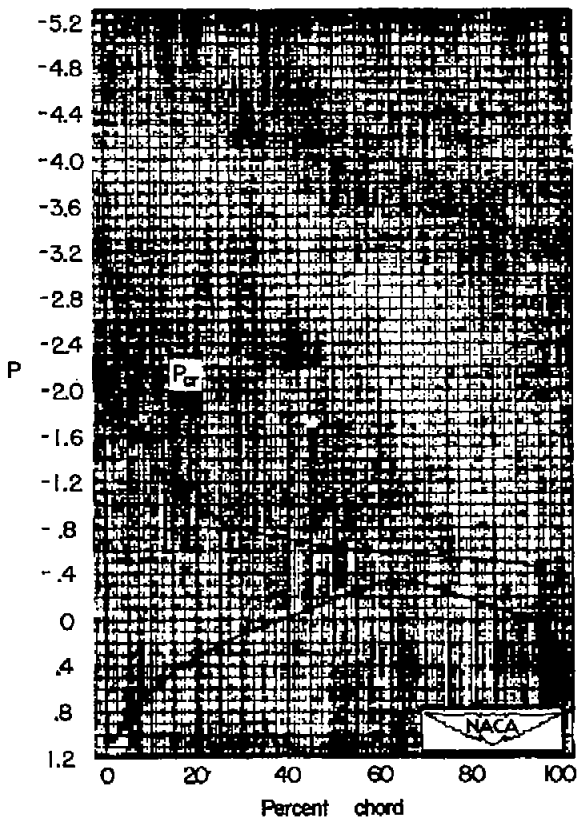
Figure 9.- Continued. M = 0.50.



Station 1



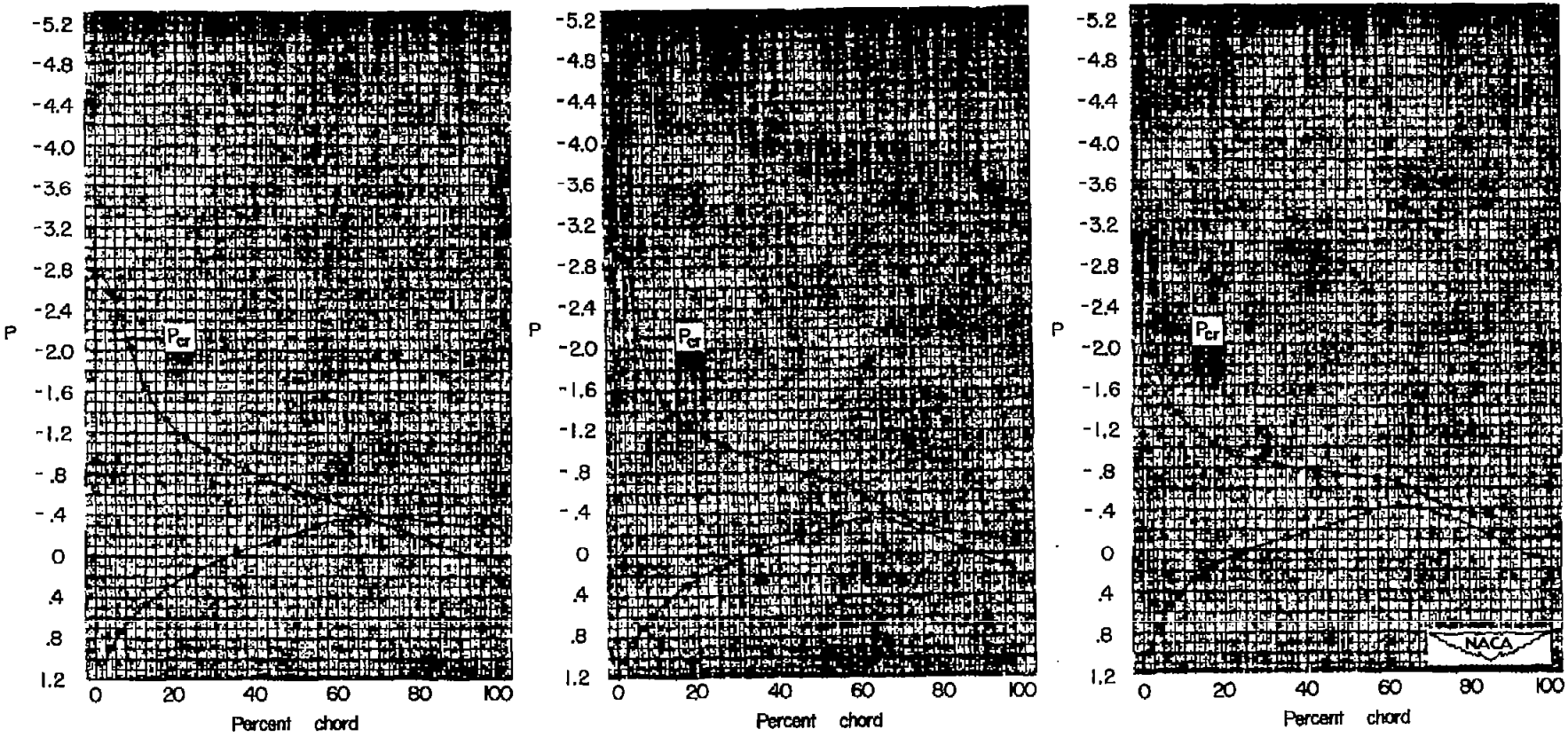
Station 2



Station 3

(e) $\alpha = 15.19^{\circ}$.

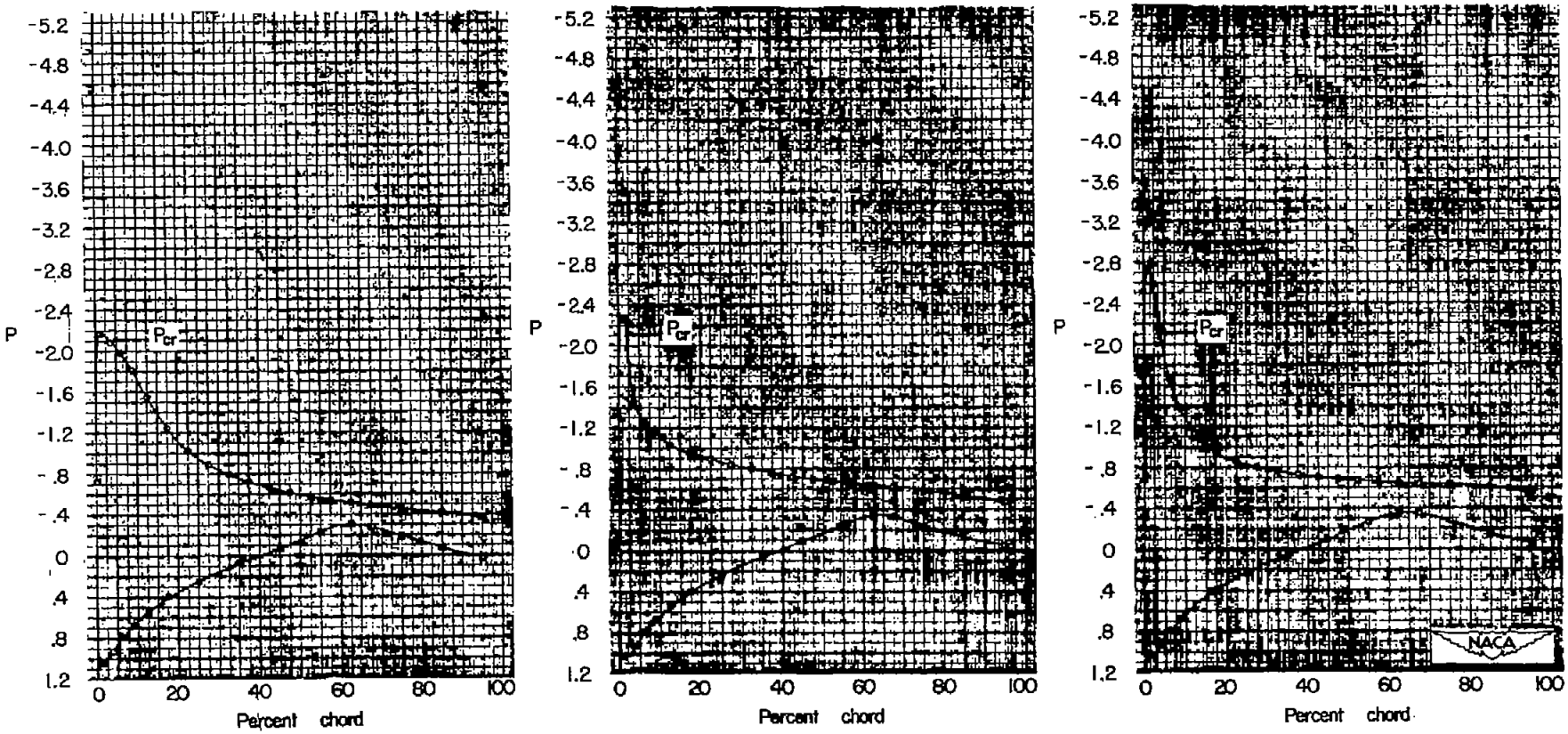
Figure 9.- Continued. M = 0.50.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 15.19^{\circ}$.

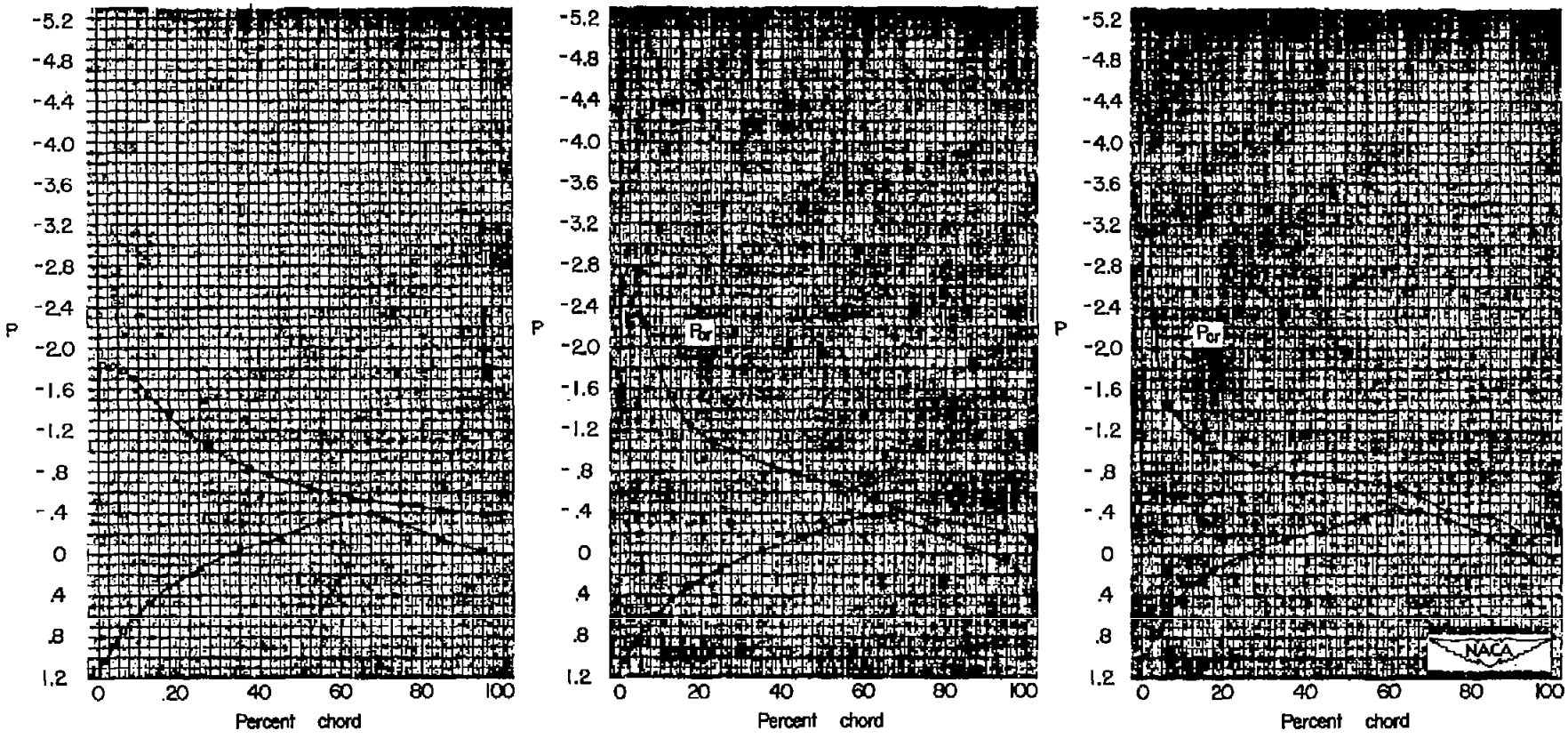
Figure 9.- Continued. M = 0.50.



Station 1 Station 2 Station 3

(f) $\alpha = 16.18^{\circ}$.

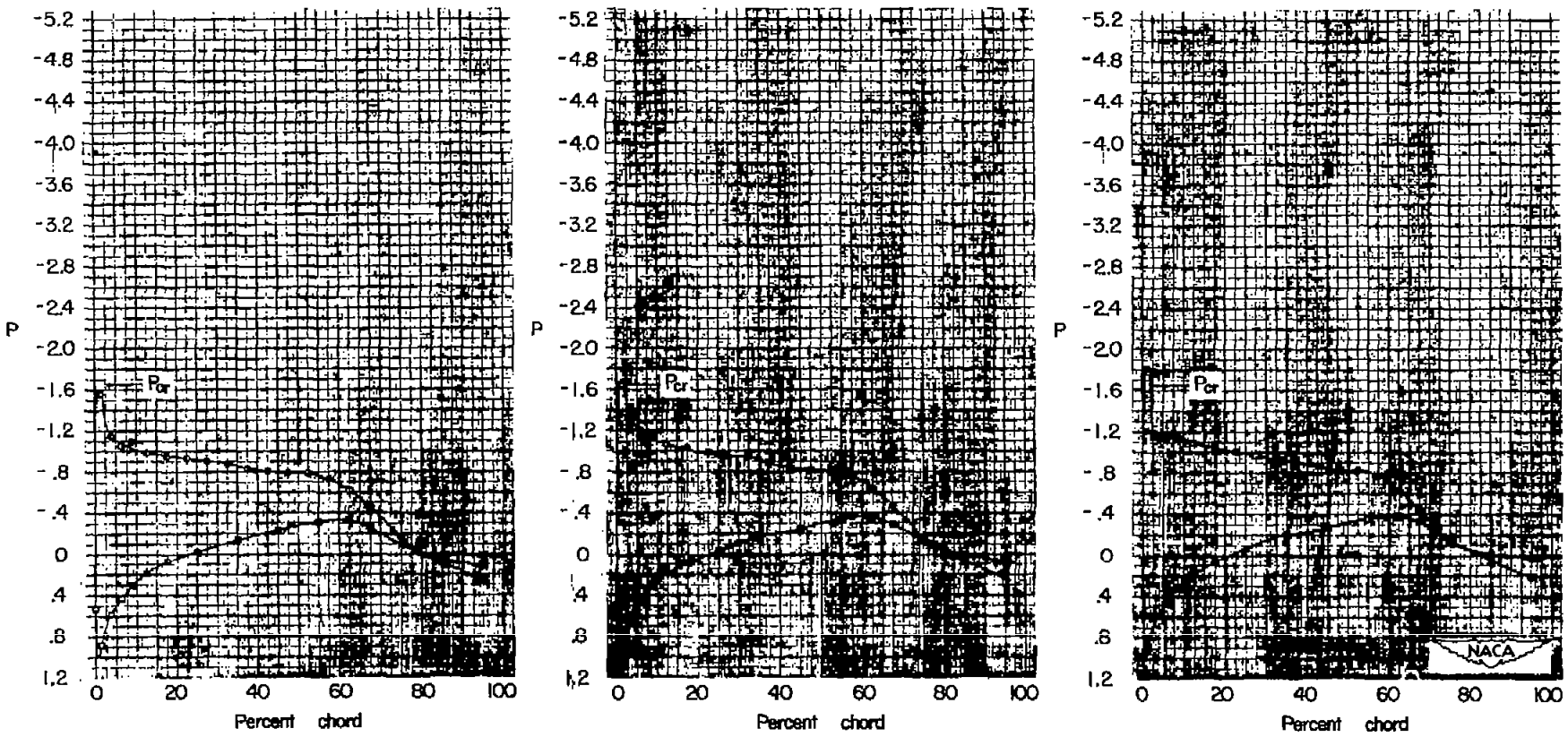
Figure 9.- Continued. M = 0.50.



Station 4 Station 5 Station 6

(f) Concluded. $\alpha = 16.18^{\circ}$.

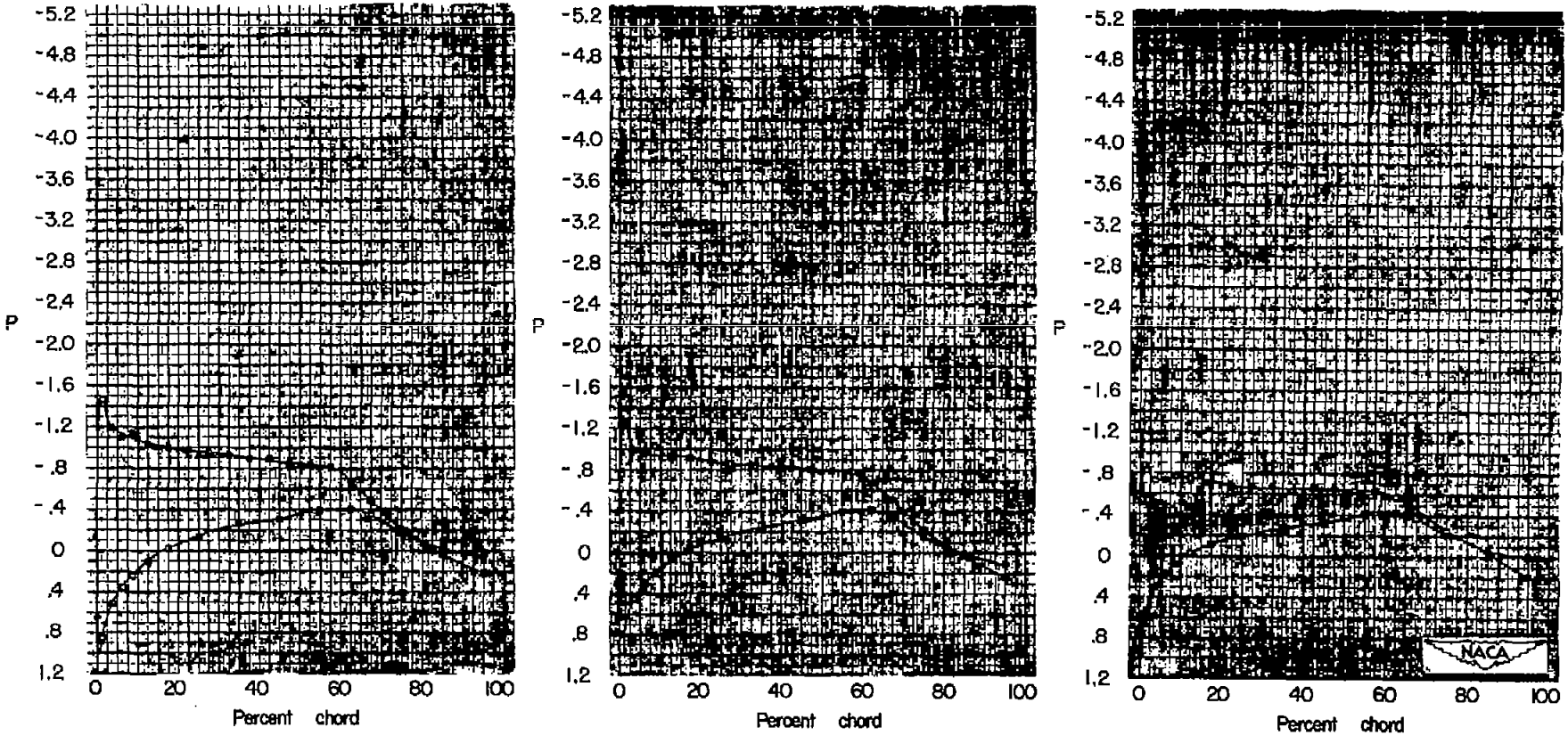
Figure 9.- Concluded. M = 0.50.



Station 1 Station 2 Station 3

(a) $\alpha = 6.81^{\circ}$.

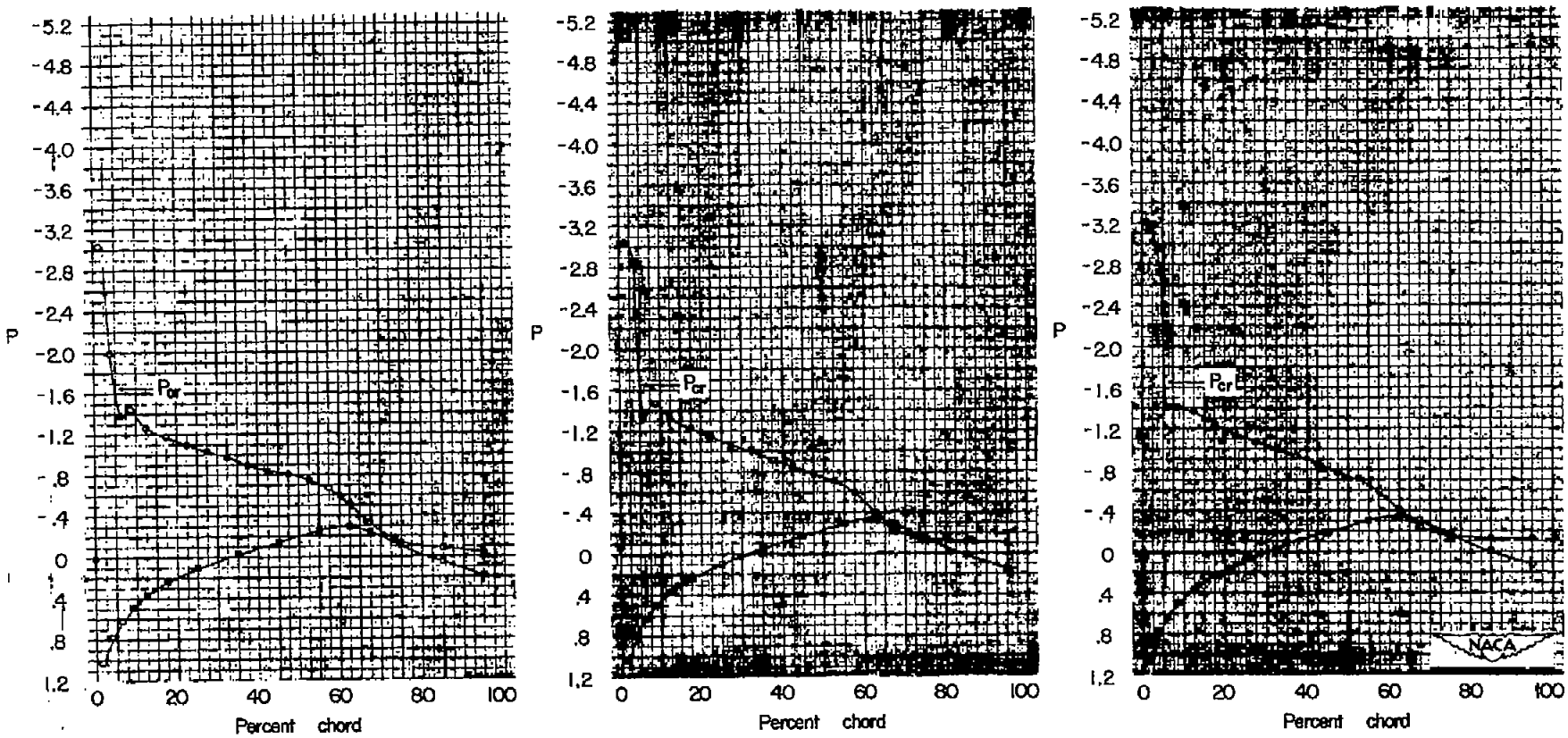
Figure 10.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.55.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = 6.81^{\circ}$.

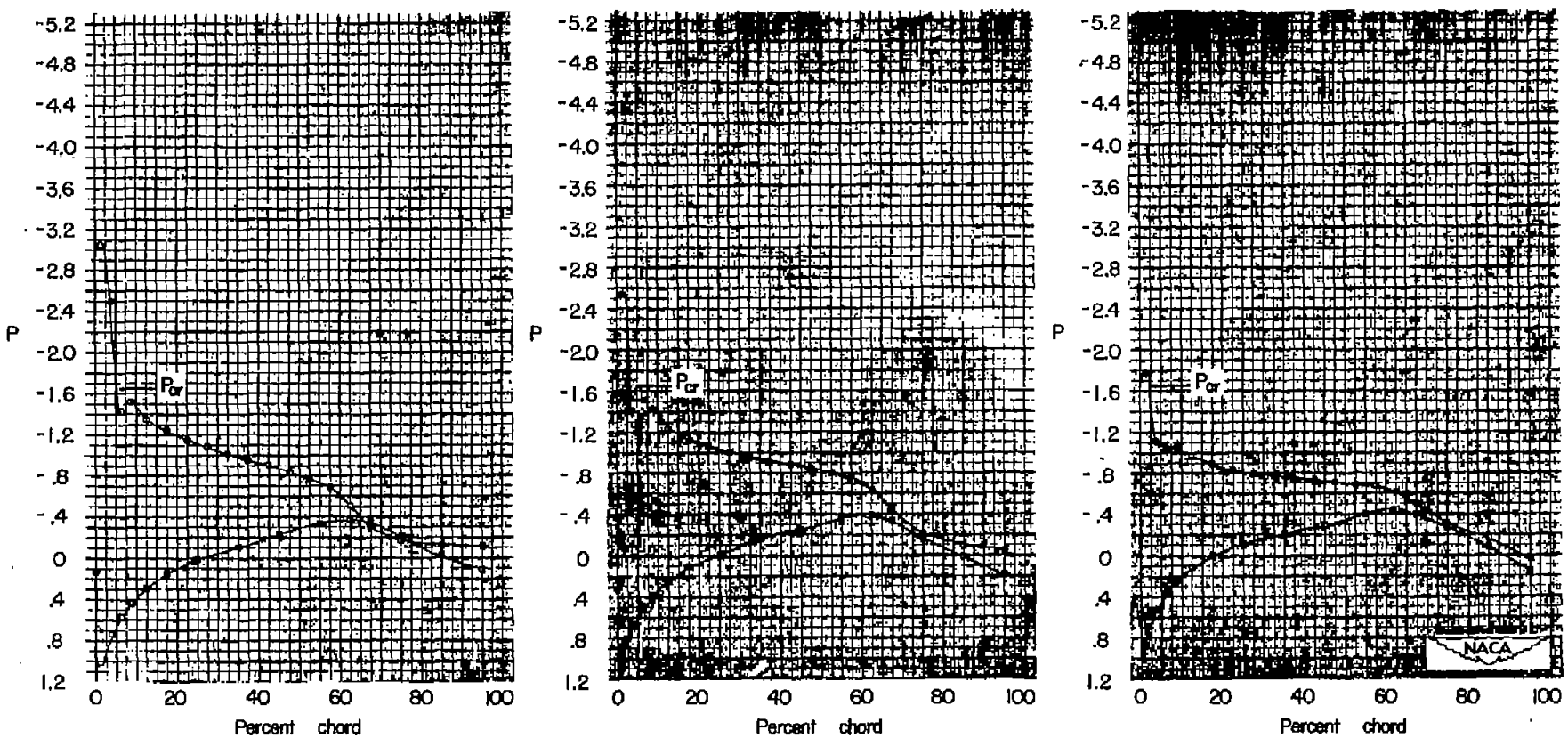
Figure 10.- Continued. M = 0.55.



Station 1 Station 2 Station 3

(b) $\alpha = 10.02^{\circ}$.

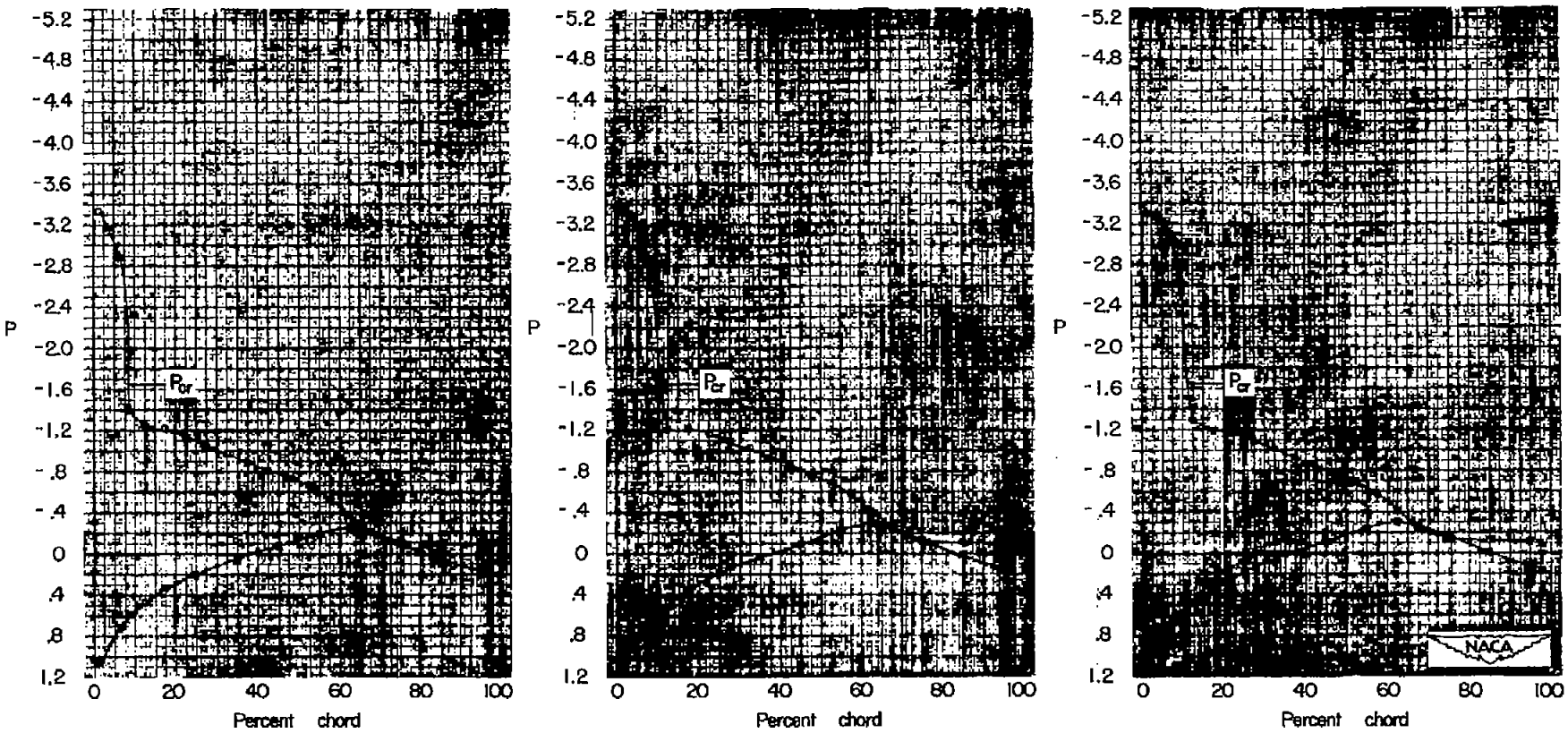
Figure 10.- Continued. M = 0.55.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = 10.02^{\circ}$.

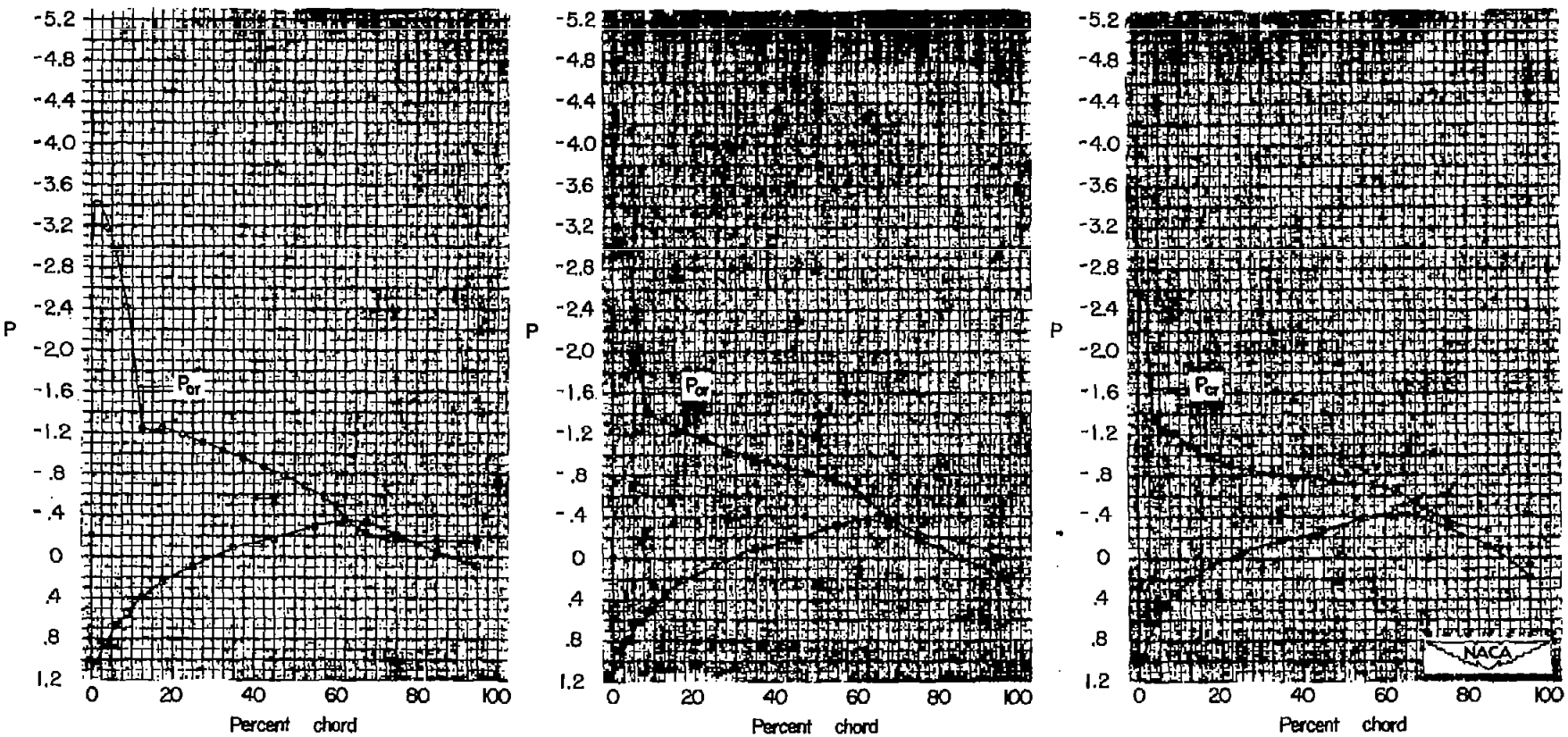
Figure 10.- Continued. M = 0.55.



Station 1 Station 2 Station 3

(c) $\alpha = 12.29^{\circ}$.

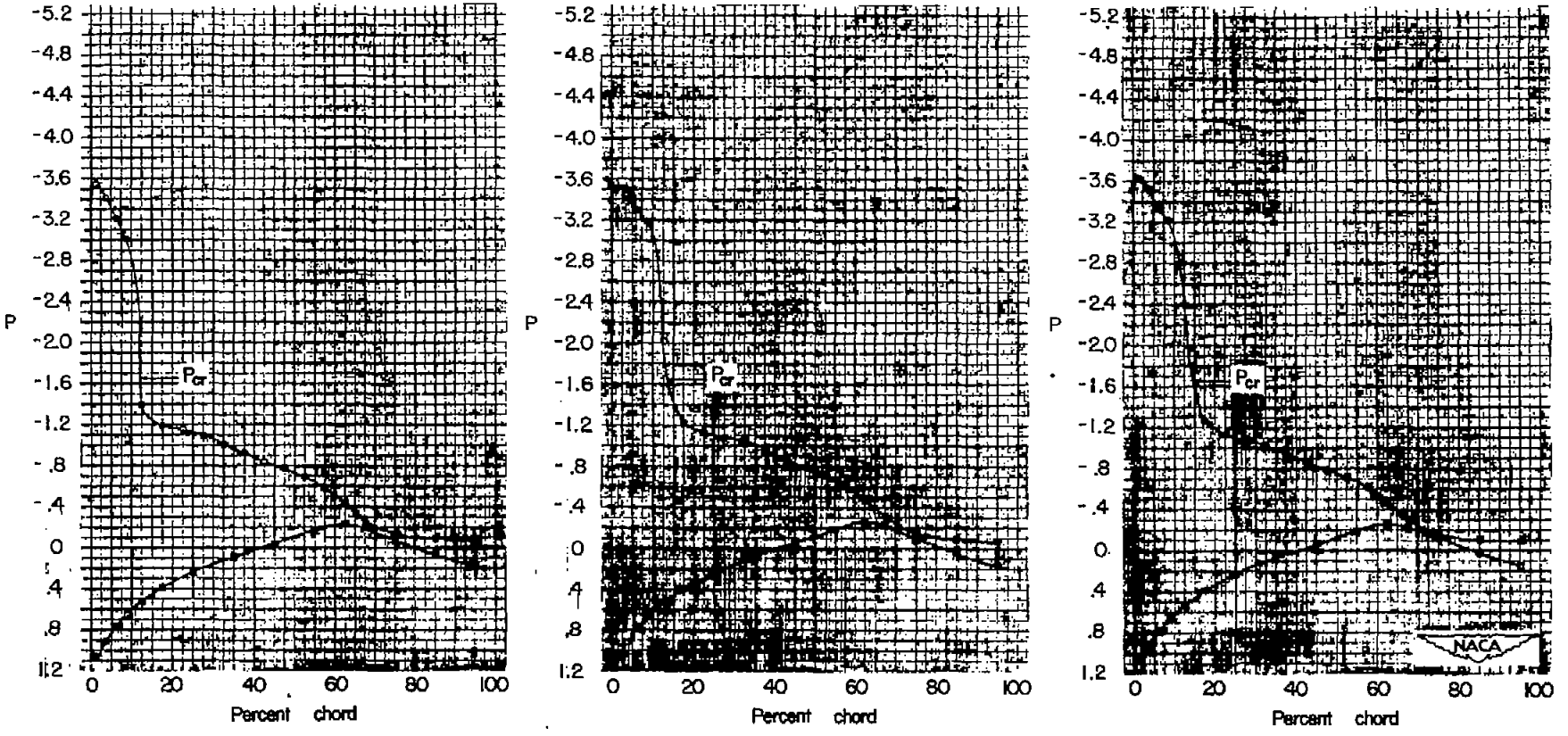
Figure 10.- Continued. M = 0.55.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 12.29^{\circ}$.

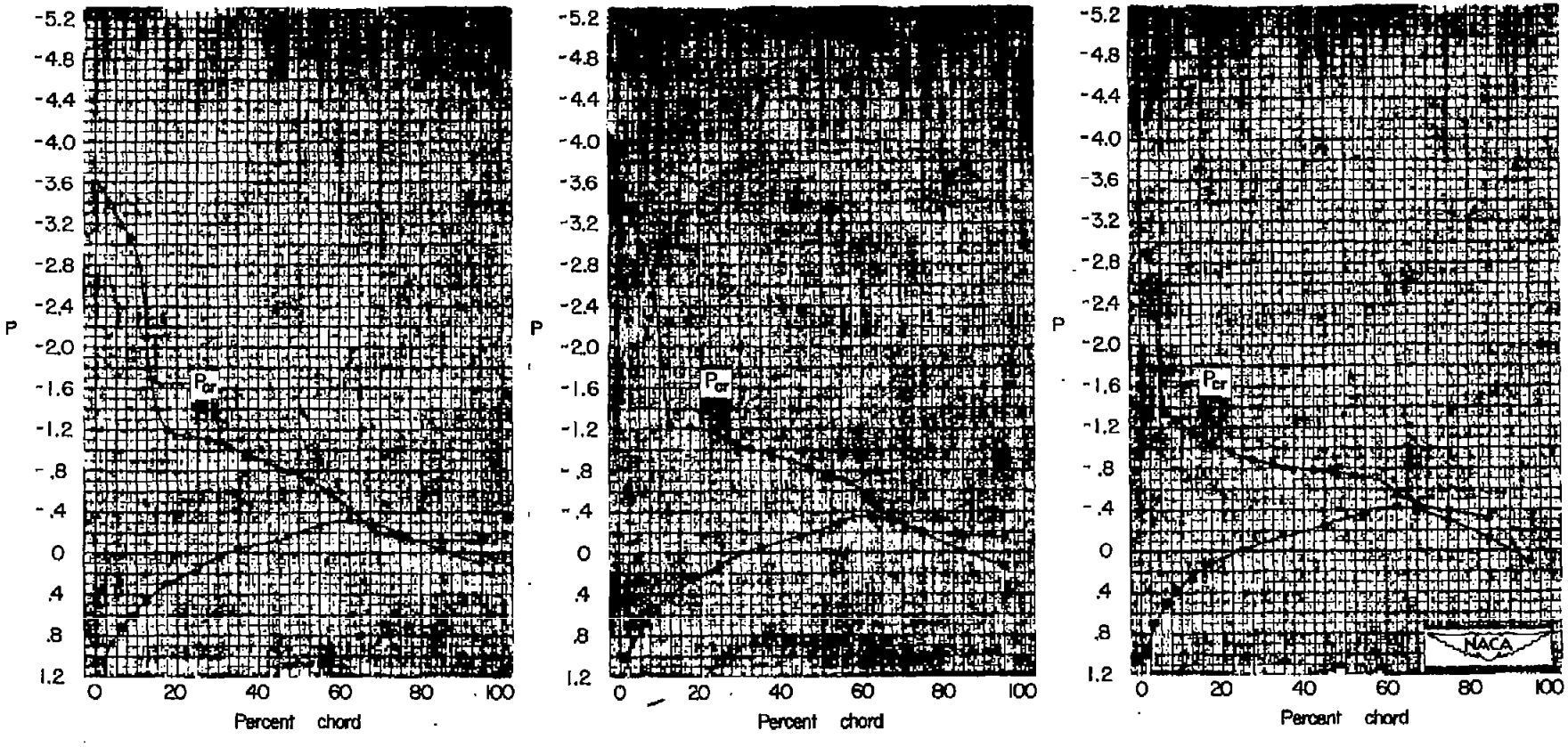
Figure 10.- Continued. M = 0.55.



Station 1 Station 2 Station 3

(d) $\alpha = 13.31^{\circ}$.

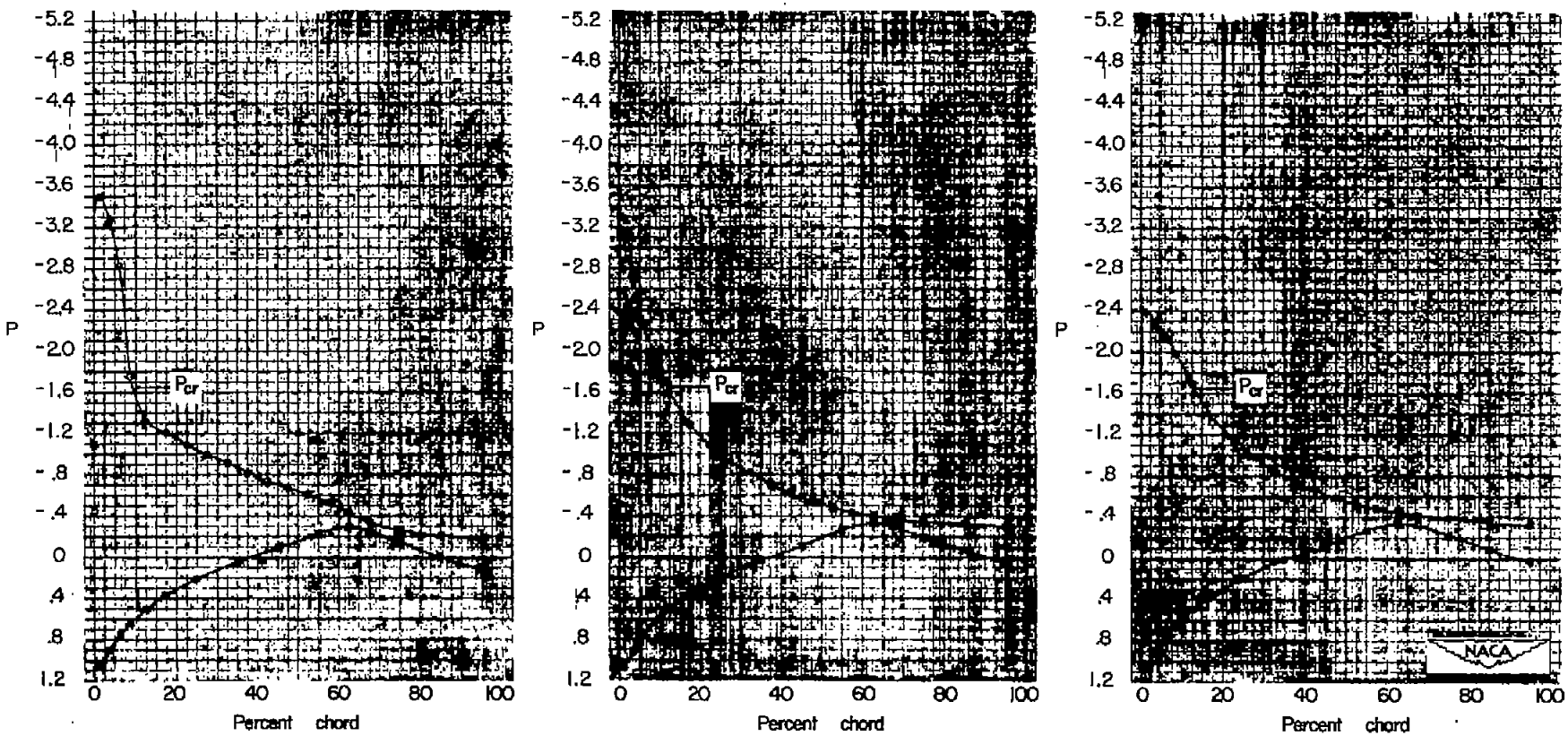
Figure 10.- Continued. M = 0.55.



Station 4 Station 5 Station 6

(d) Concluded. $\alpha = 13.31^{\circ}$.

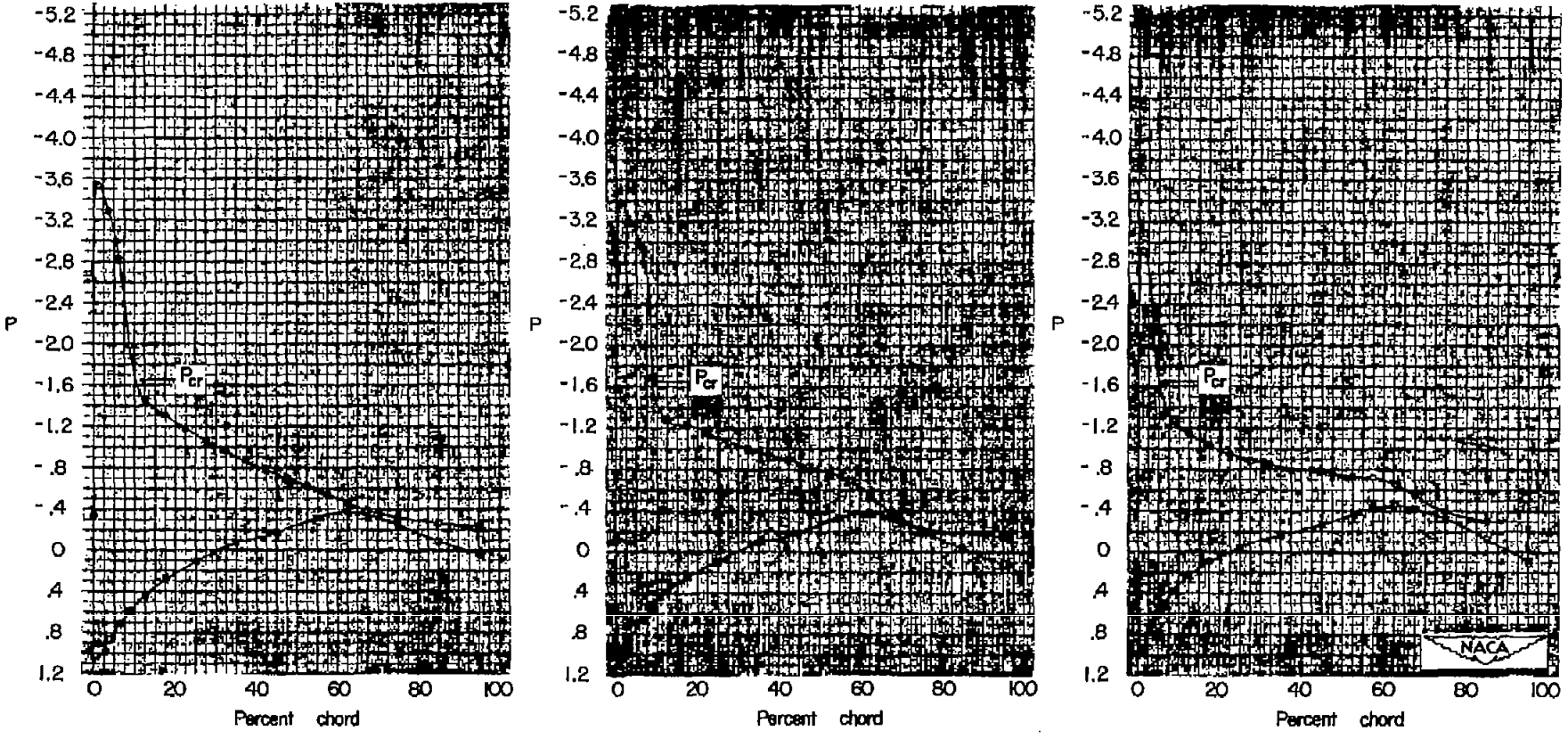
Figure 10.- Continued. M = 0.55.



Station 1 Station 2 Station 3

(e) $\alpha = 13.96^{\circ}$.

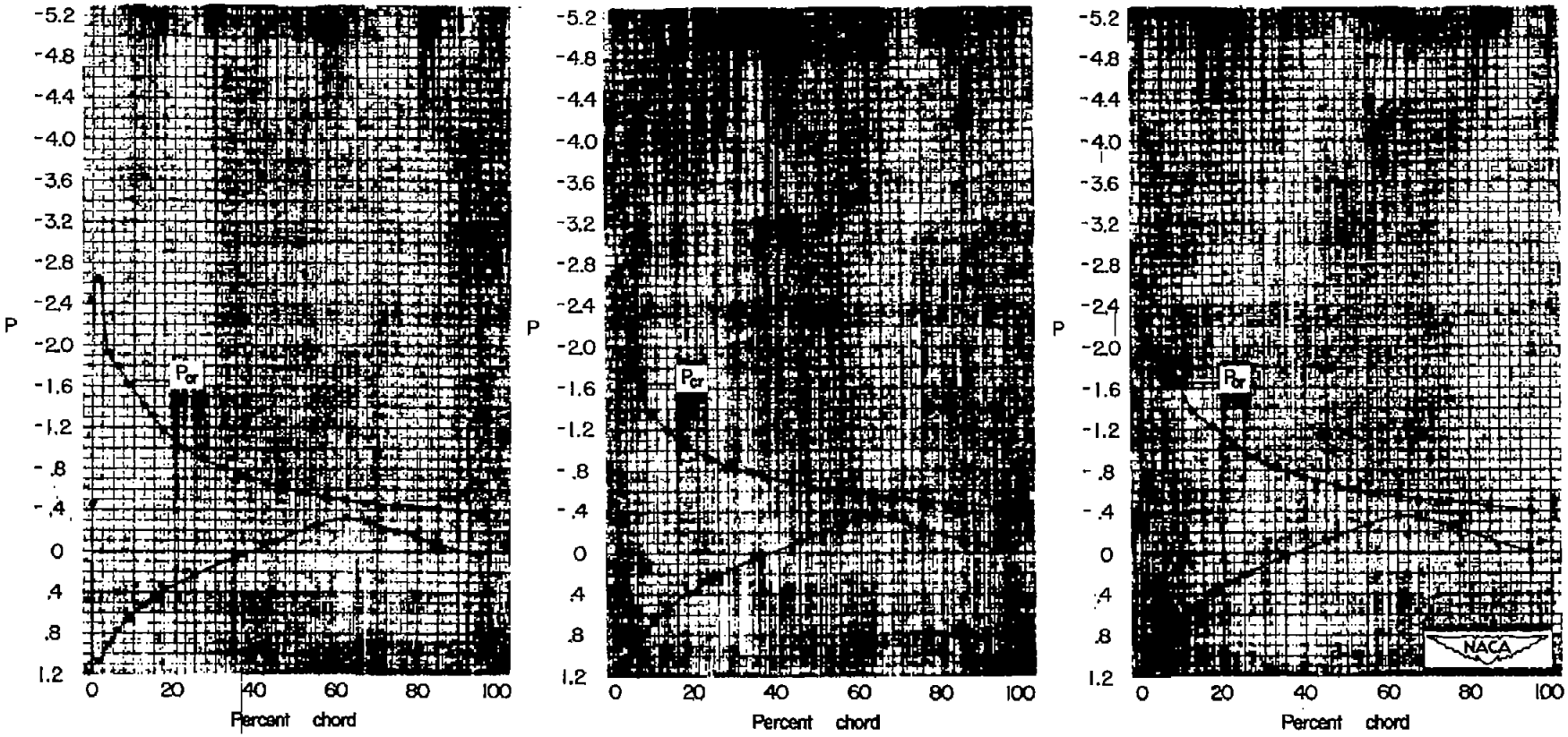
Figure 10.- Continued. M = 0.55.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 13.96^{\circ}$.

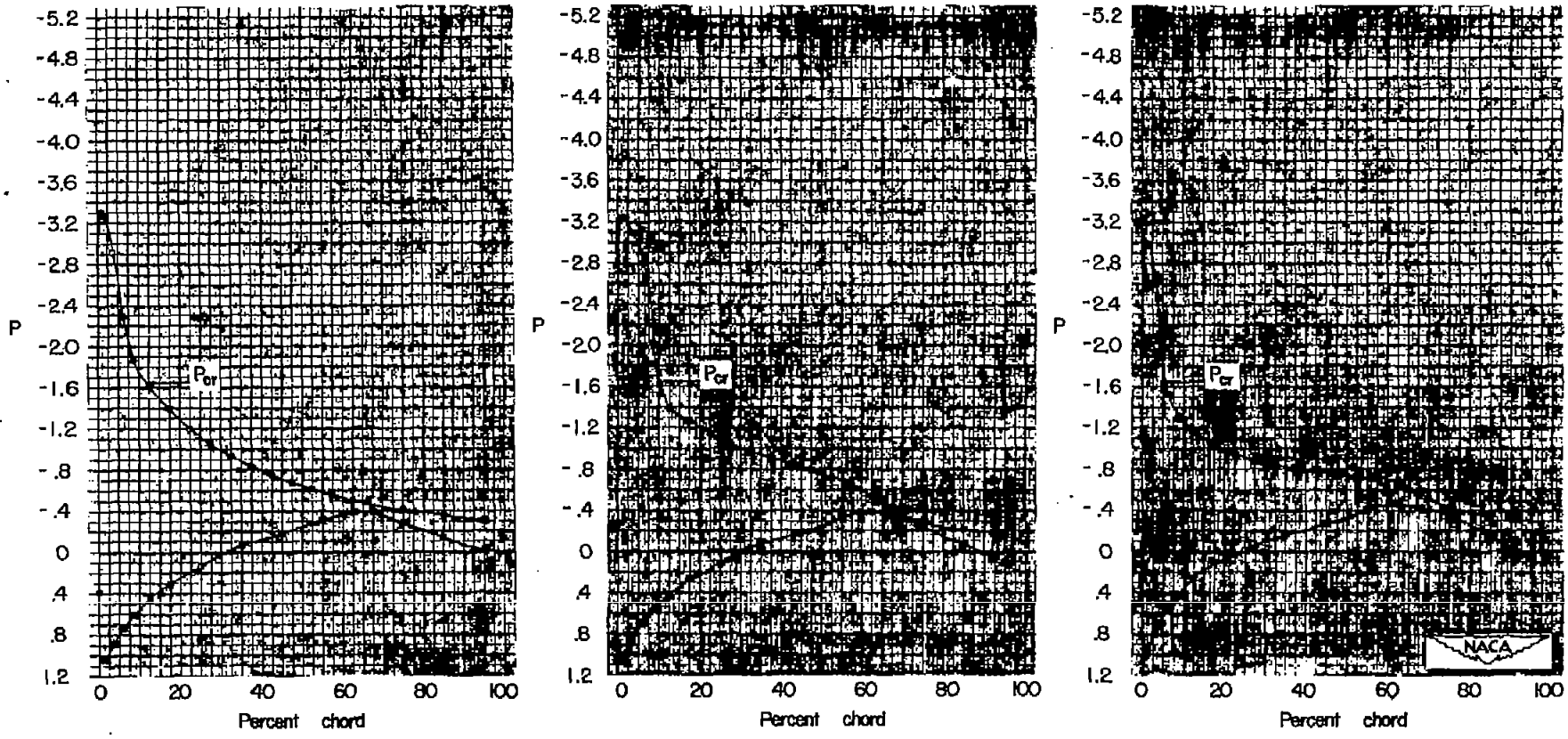
Figure 10.- Continued. M = 0.55.



Station 1 Station 2 Station 3

(f) $\alpha = 15.45^{\circ}$.

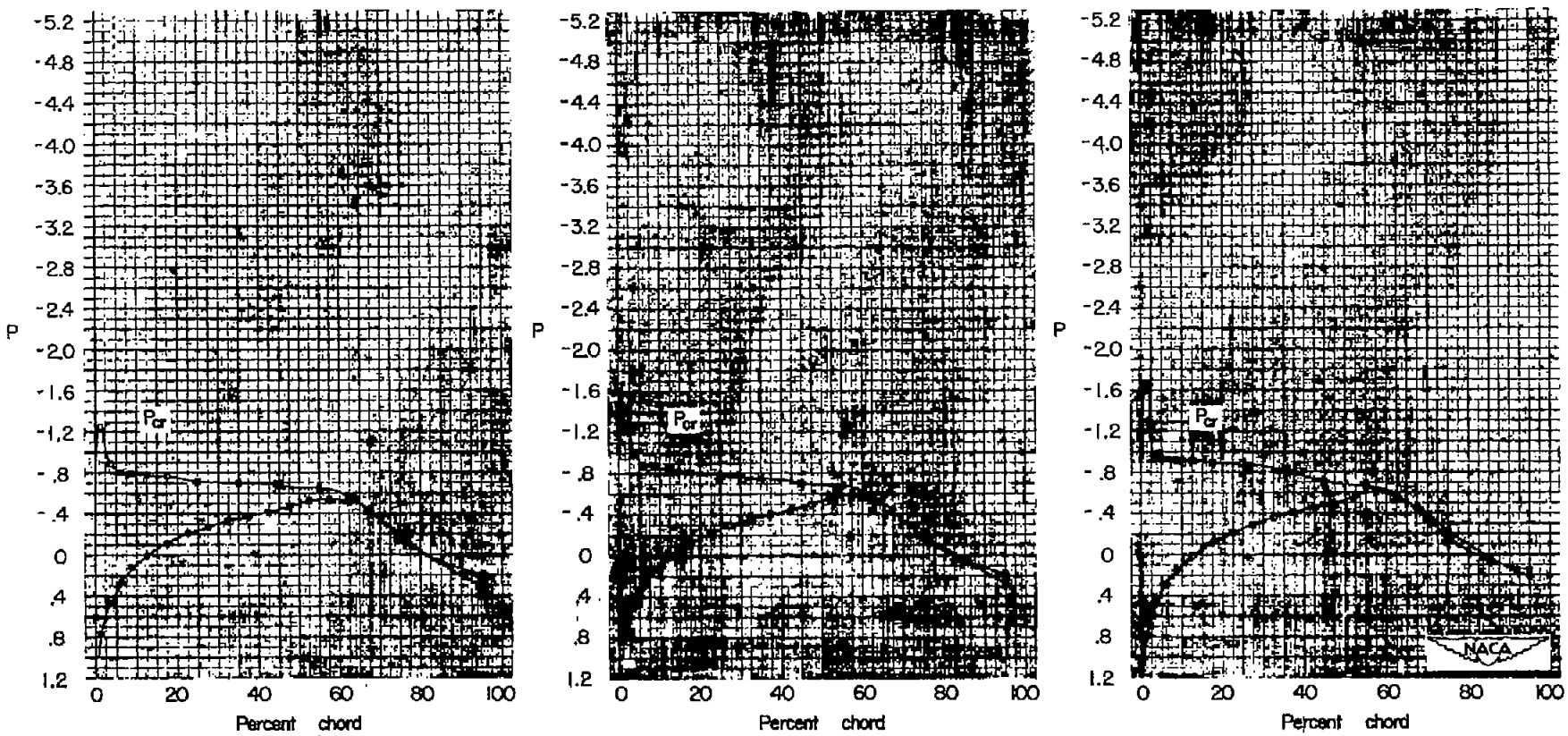
Figure 10.- Continued. M = 0.55.



Station 4 Station 5 Station 6

(f) Concluded. $\alpha = 15.45^{\circ}$.

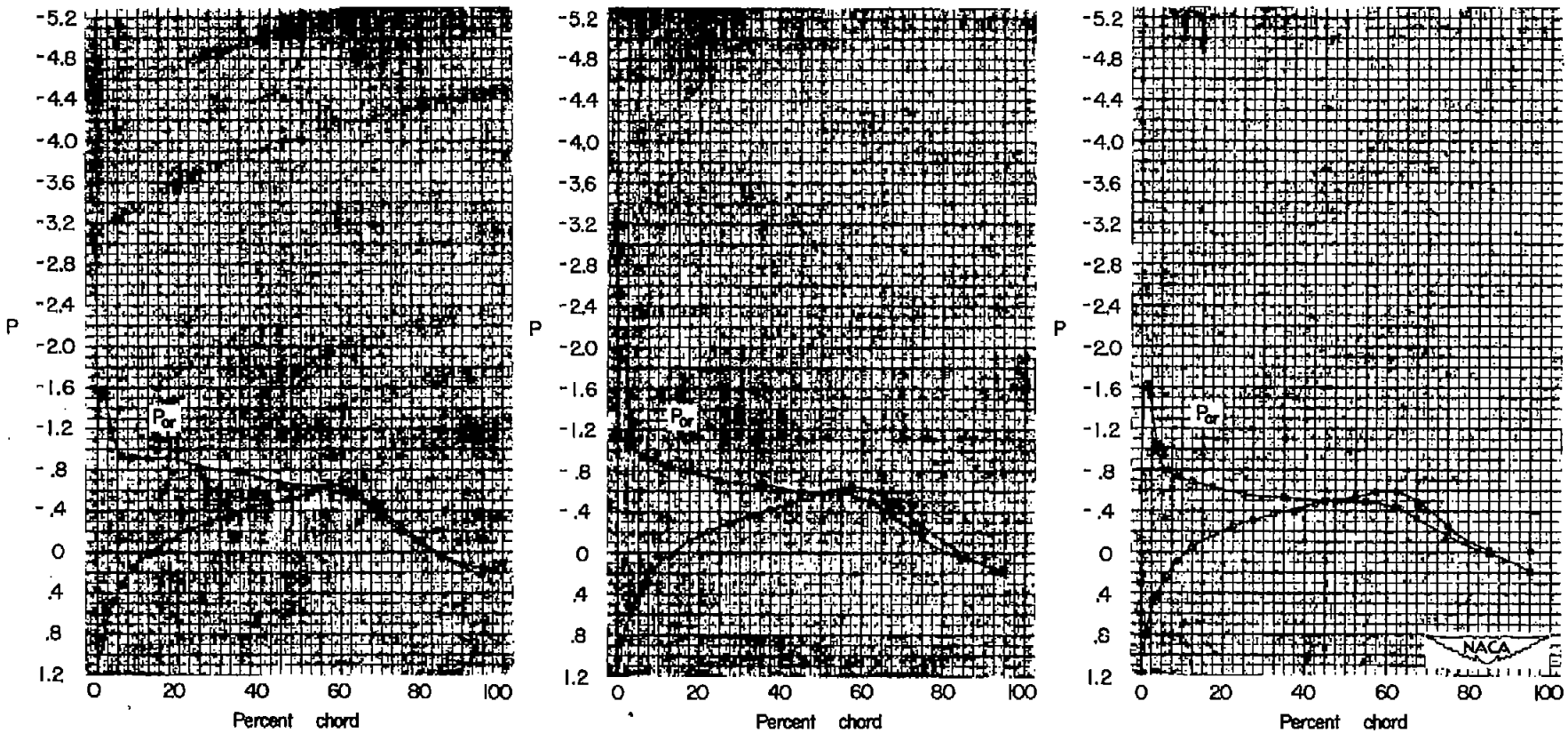
Figure 10.- Concluded. M = 0.55.



Station 1 Station 2 Station 3

(a) $\alpha = -4.13^{\circ}$.

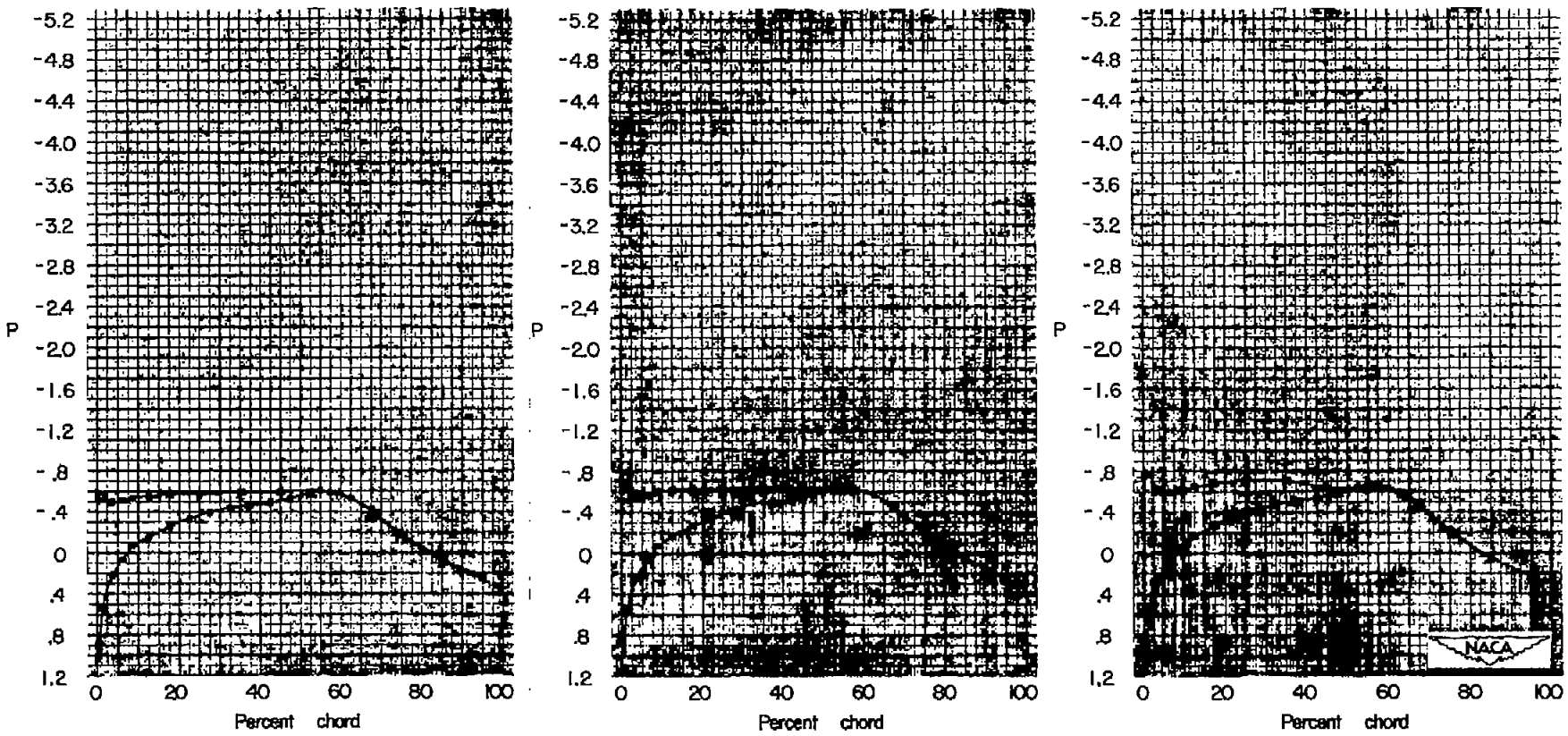
Figure 11.- Experimental pressure distribution obtained on a wing of the NACA 66-series airfoil sections. M = 0.60.



Station 4 Station 5 Station 6

(a) Concluded. $\alpha = -4.13^{\circ}$.

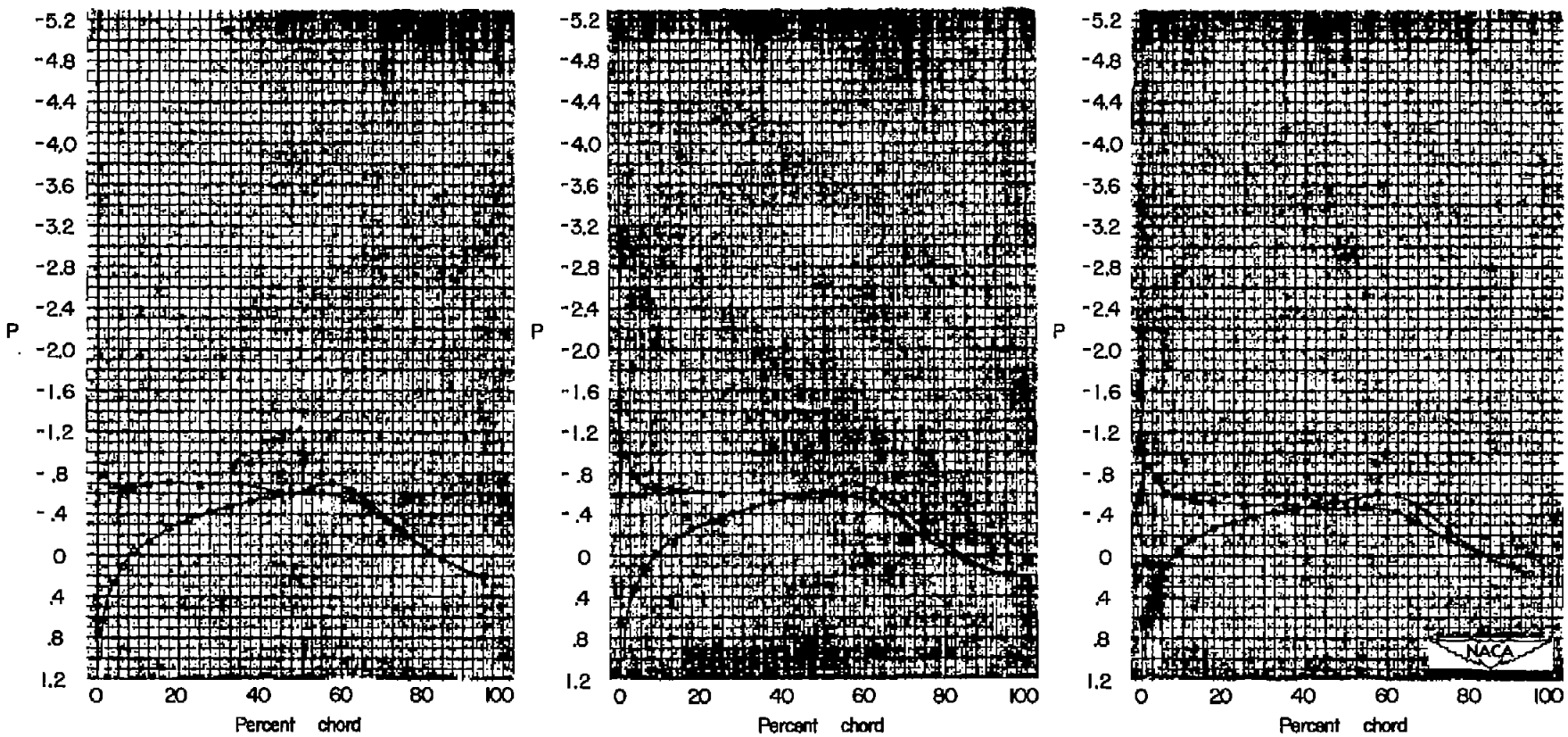
Figure 11.- Continued. M = 0.60.



Station 1 Station 2 Station 3

(b) $\alpha = -1.95$.

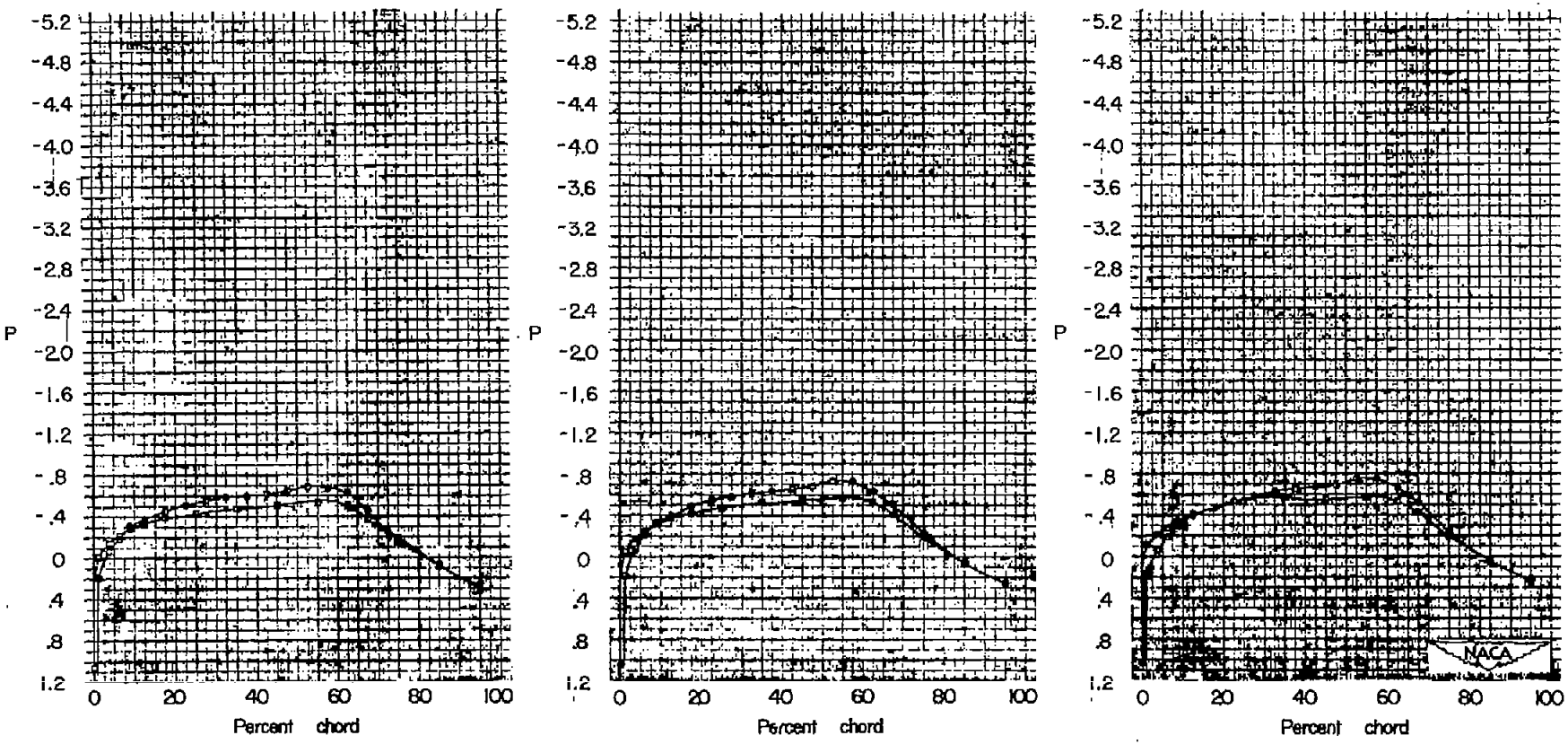
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(b) Concluded. $\alpha = -1.95^{\circ}$.

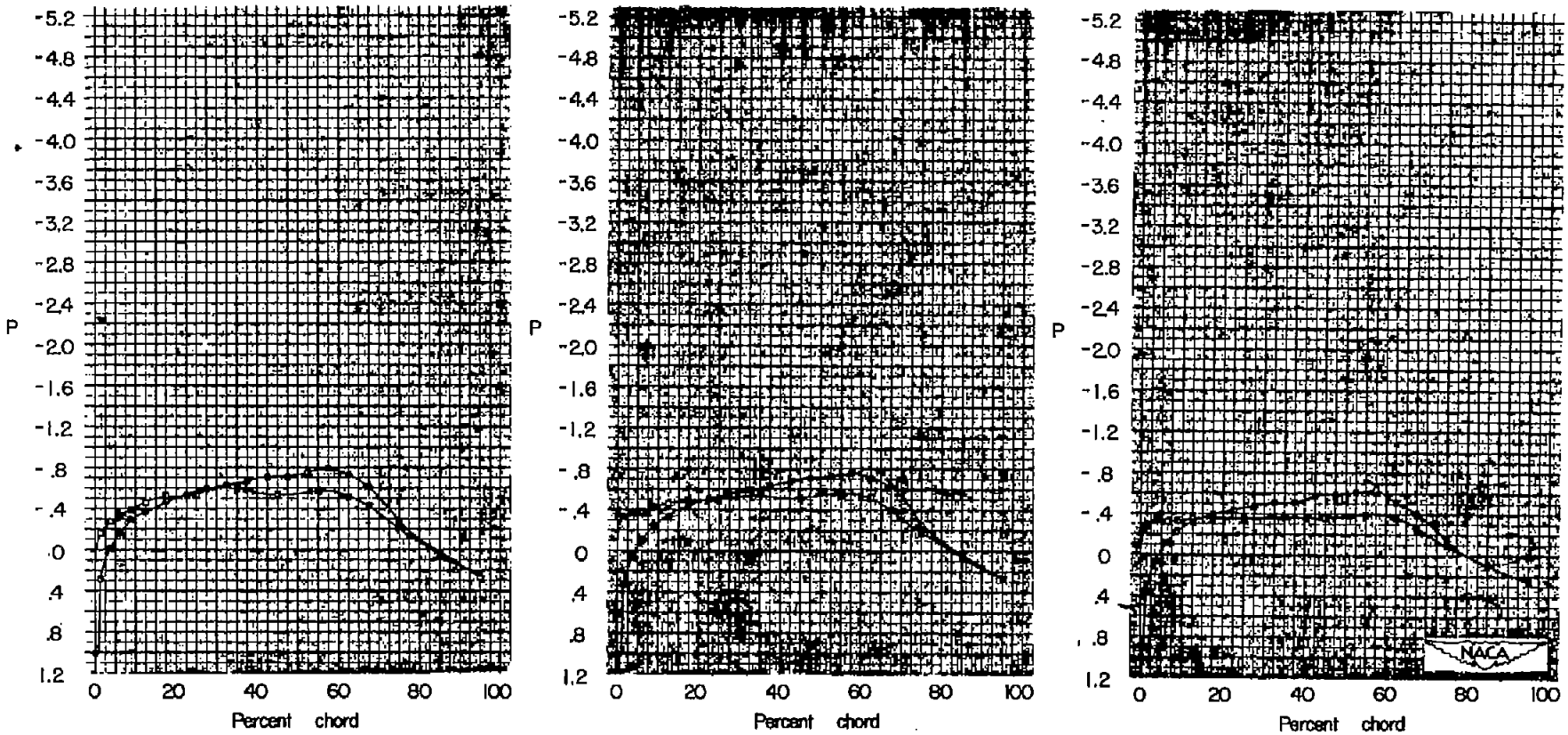
Figure 11.- Continued. M = 0.60.



Station 1 Station 2 Station 3

(c) $\alpha = 0.26^{\circ}$.

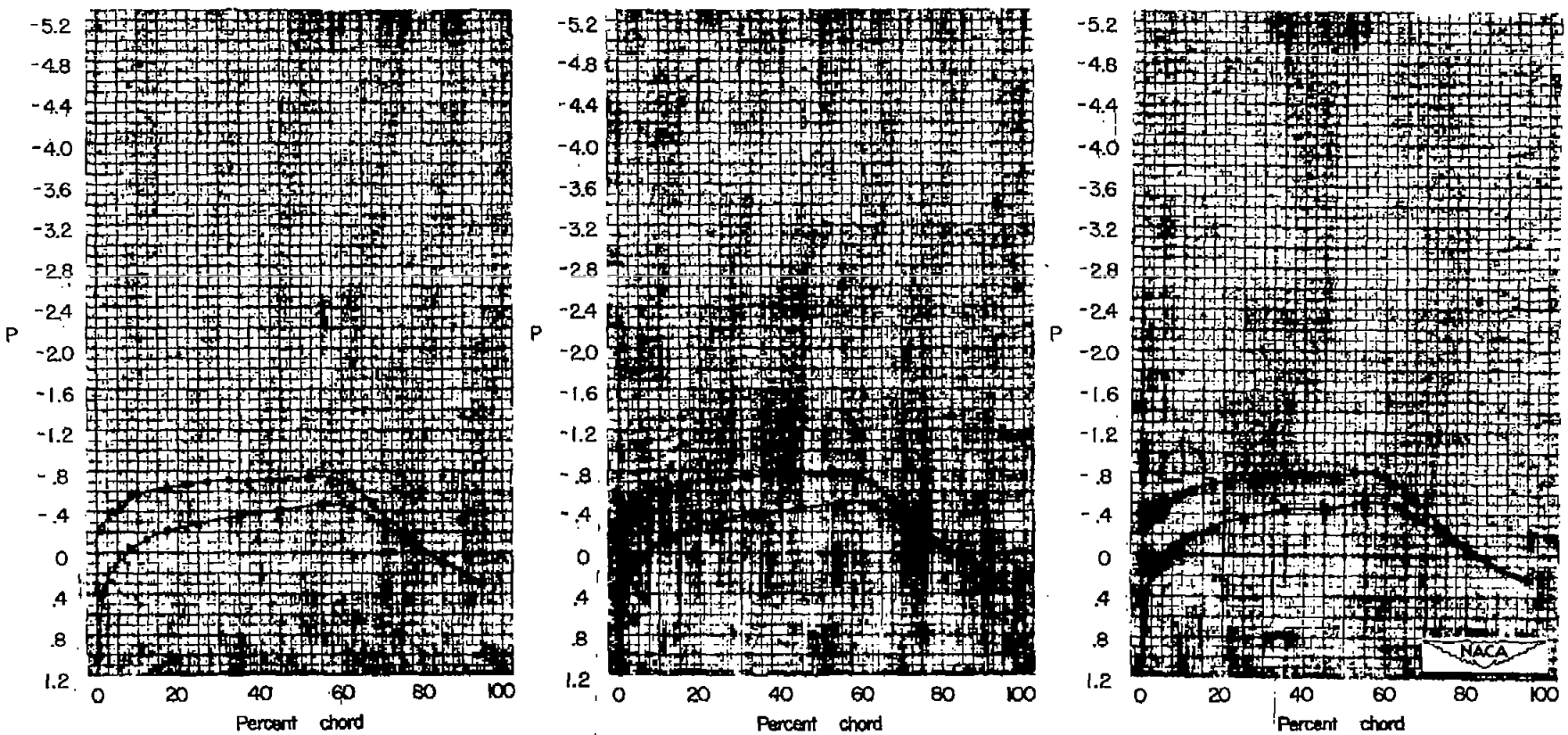
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(c) Concluded. $\alpha = 0.26^{\circ}$.

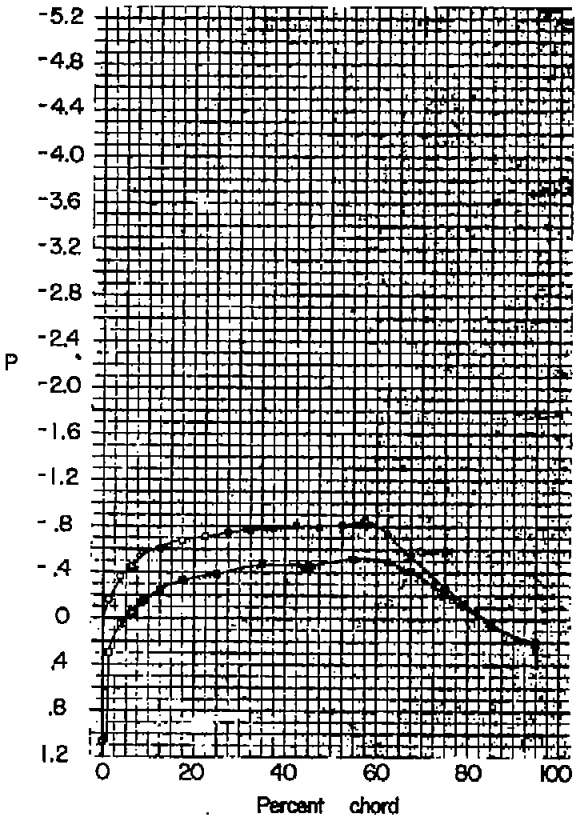
Figure 11.- Continued. M = 0.60.



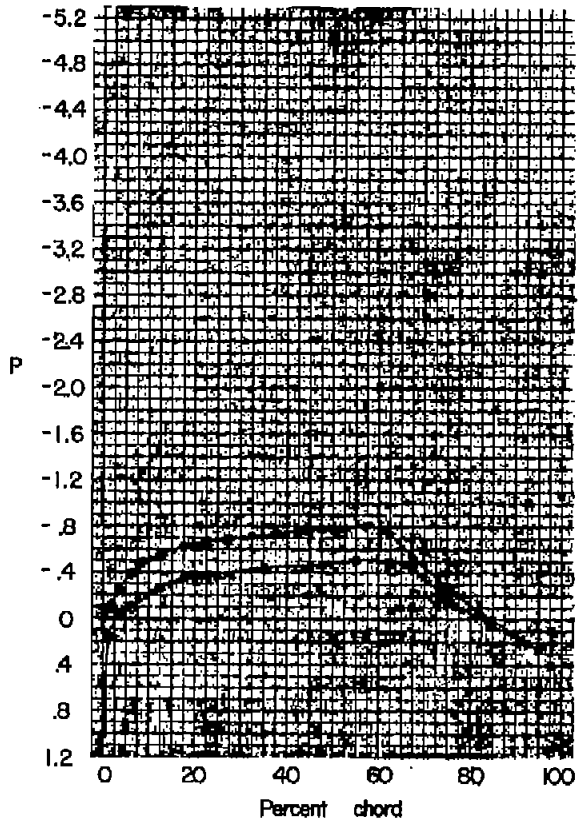
Station 1 Station 2 Station 3

(d) $\alpha = 2.46^{\circ}$.

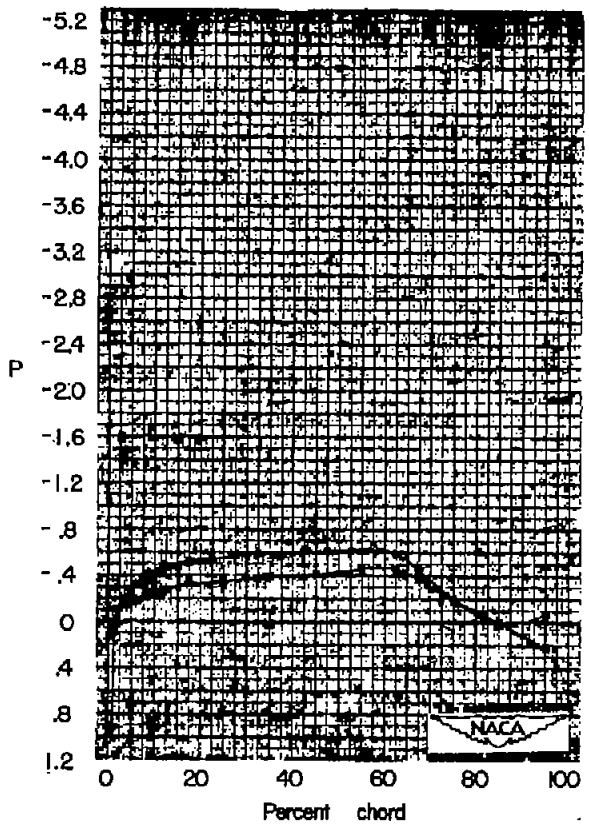
Figure 11.- Continued. M = 0.60.



Station 4



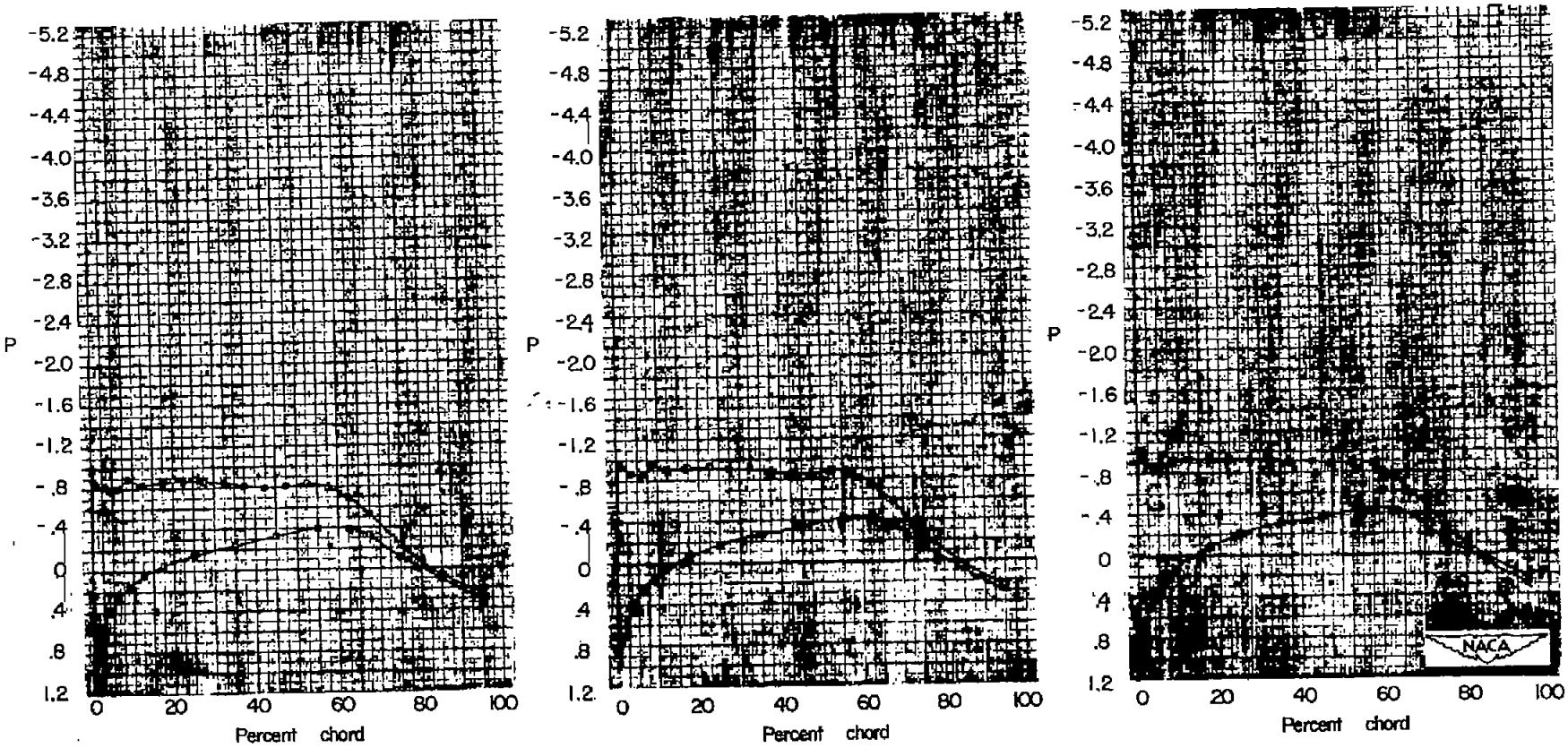
Station 5



Station 6

(d) Concluded. $\alpha = 2.46^{\circ}$.

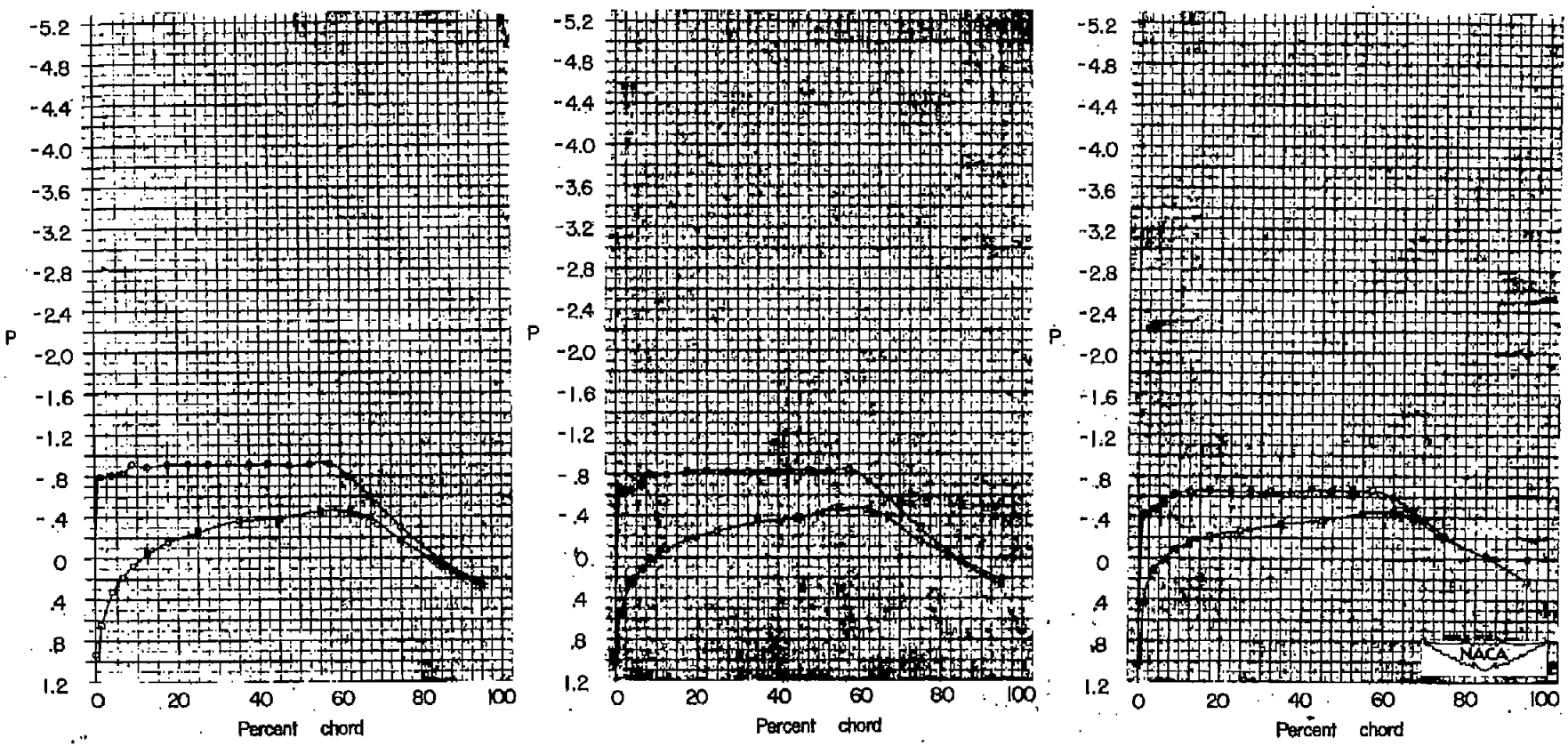
Figure 11.- Continued. M = 0.60.



Station 1 Station 2 Station 3

(e) $\alpha = 4.64^{\circ}$.

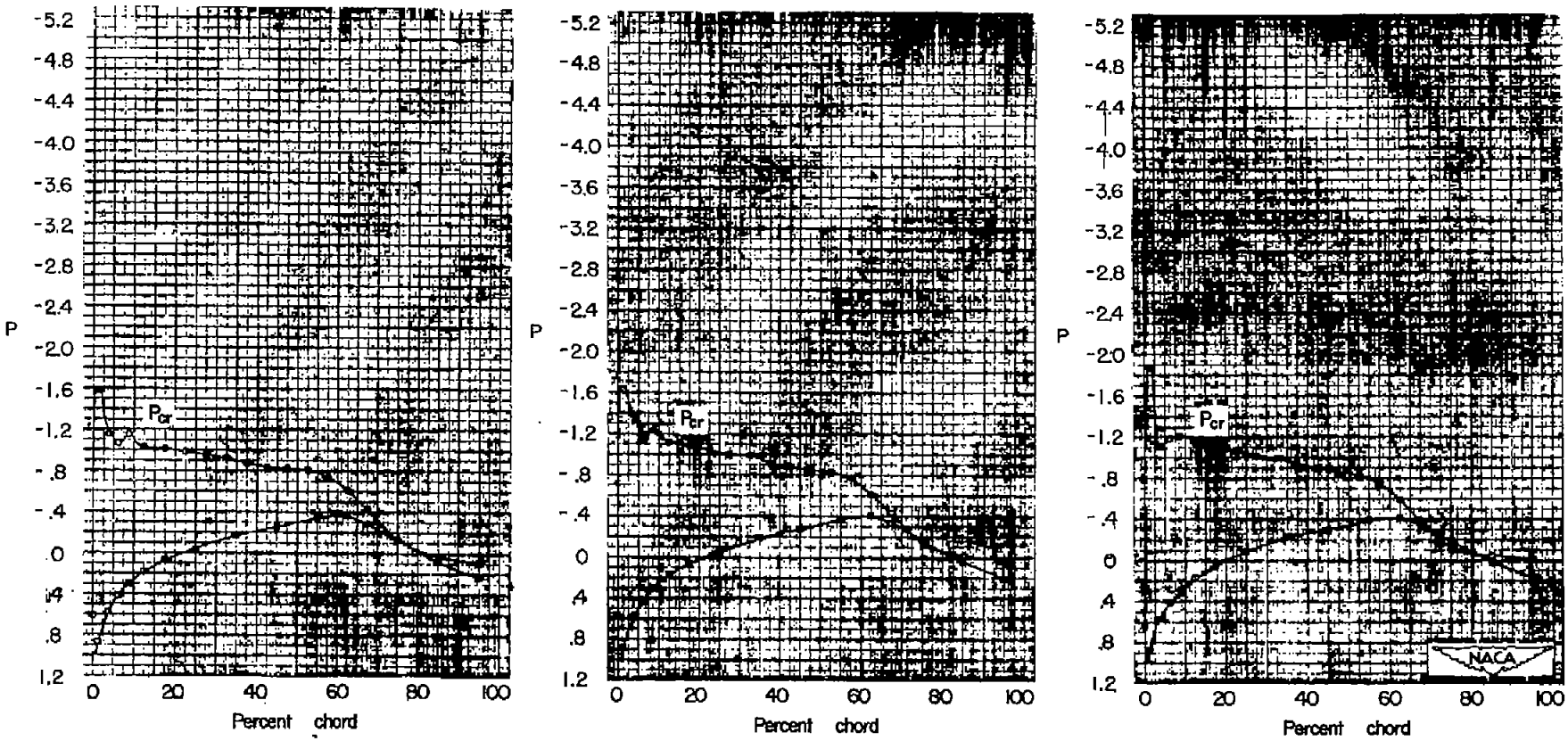
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(e) Concluded. $\alpha = 4.64^{\circ}$.

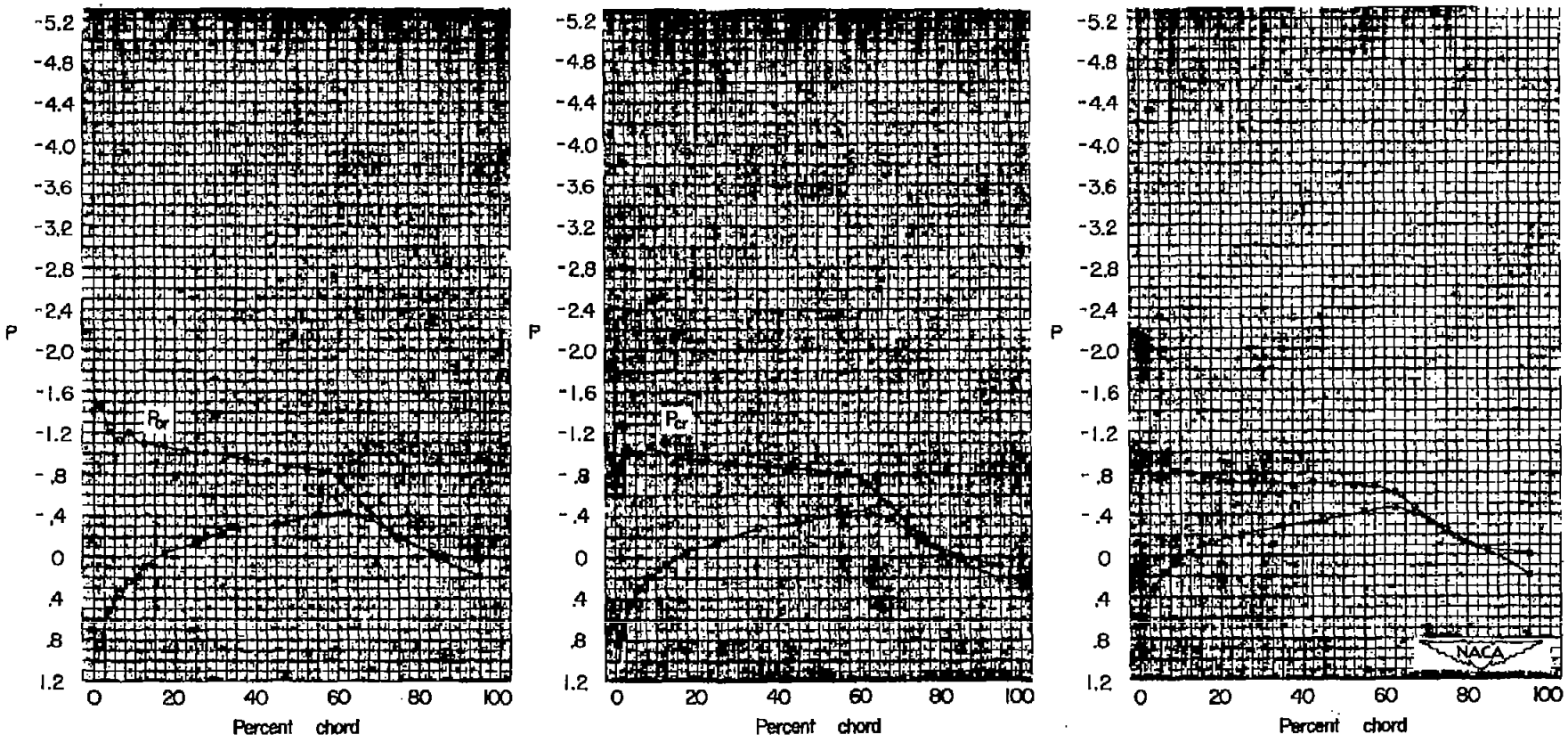
Figure 11.- Continued. M = 0.60.



Station 1 Station 2 Station 3

(f) $\alpha = 6.81^{\circ}$.

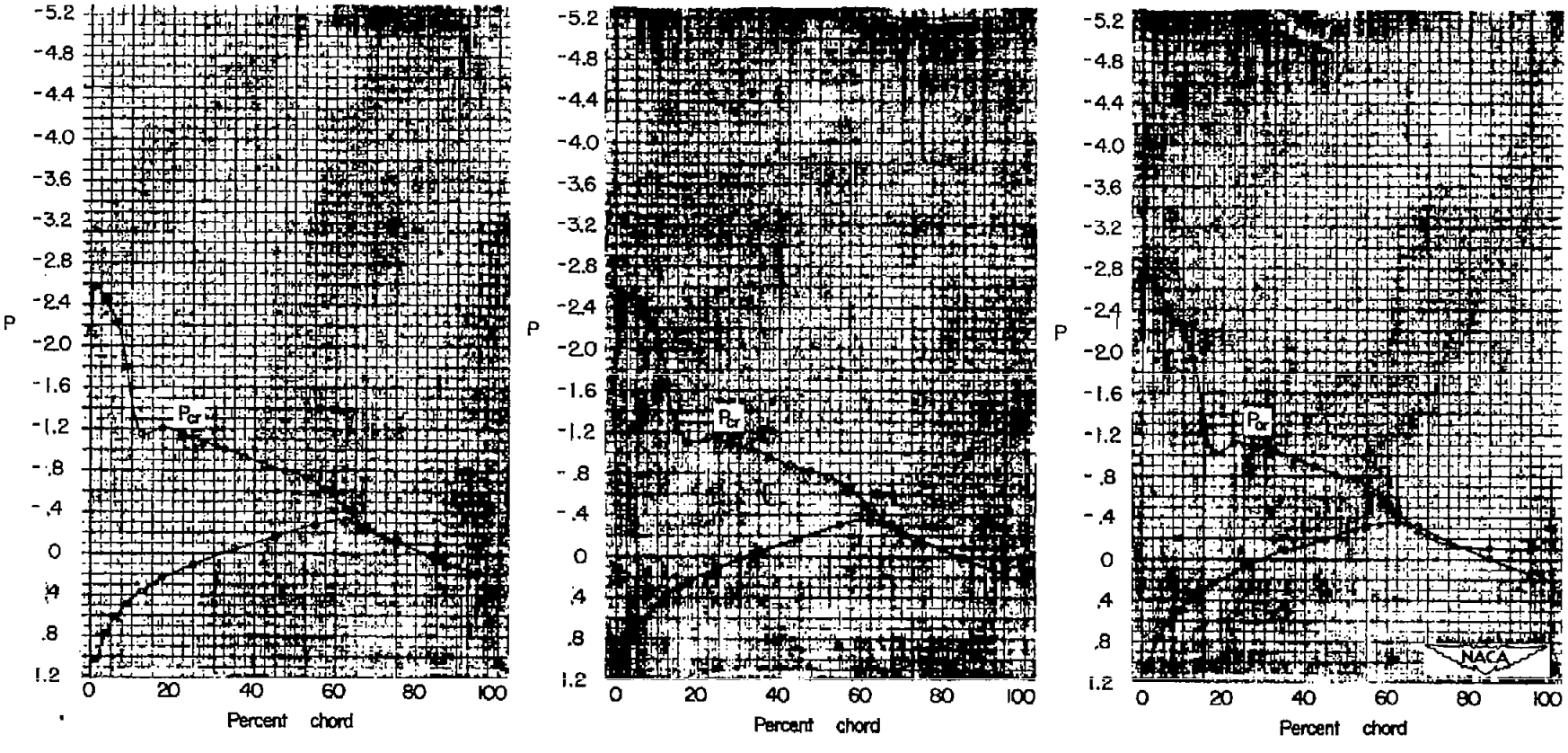
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(f) Concluded. $\alpha = 6.81^{\circ}$.

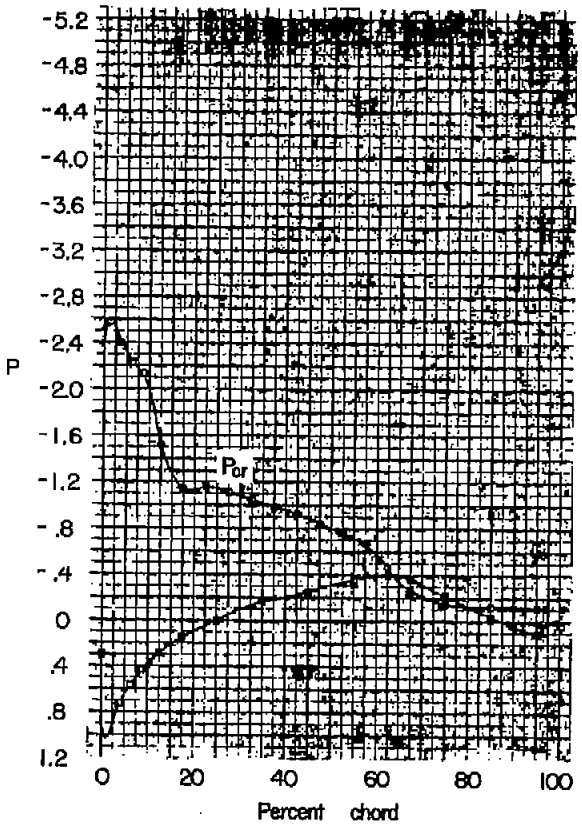
Figure 11.- Continued. M = 0.60.



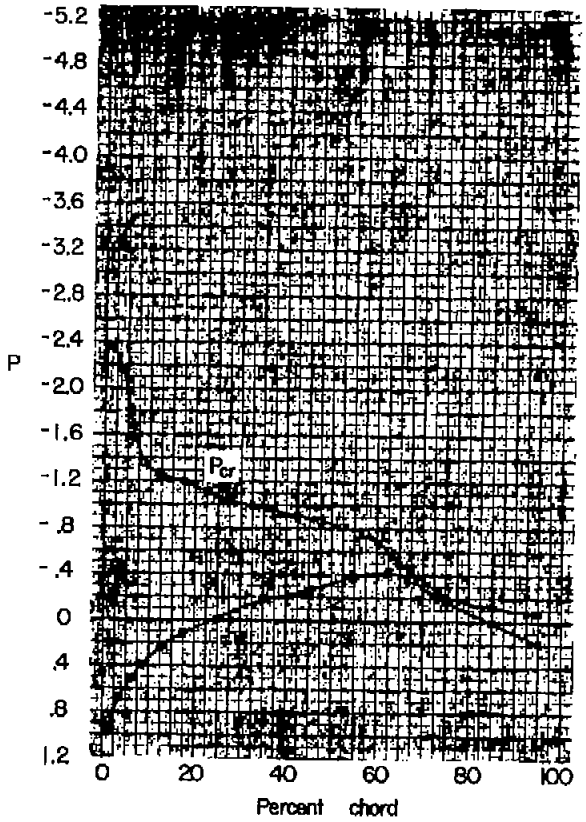
Station 1 Station 2 Station 3

(g) $\alpha = 10.05^{\circ}$.

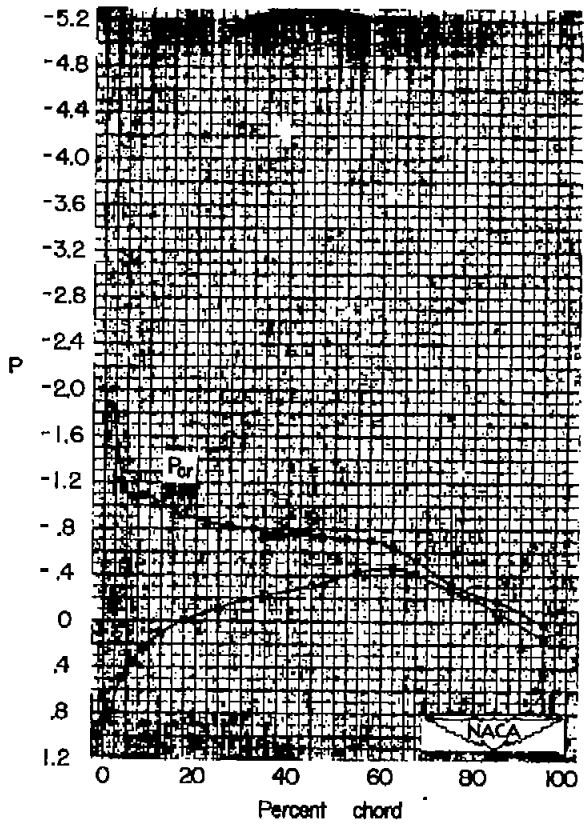
Figure 11.- Continued. M = 0.60.



Station 4



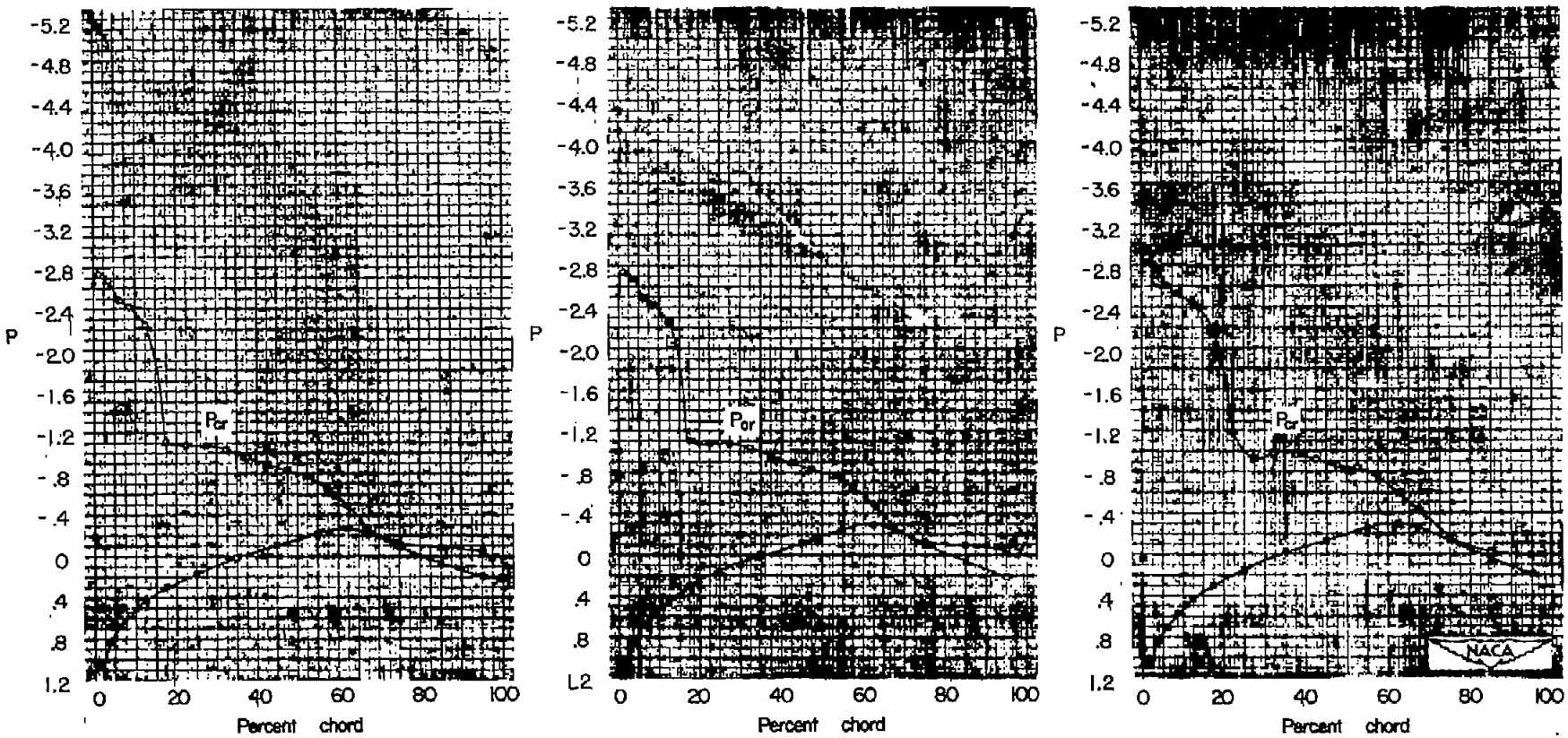
Station 5



Station 6

(g) Concluded. $\alpha = 10.05^{\circ}$.

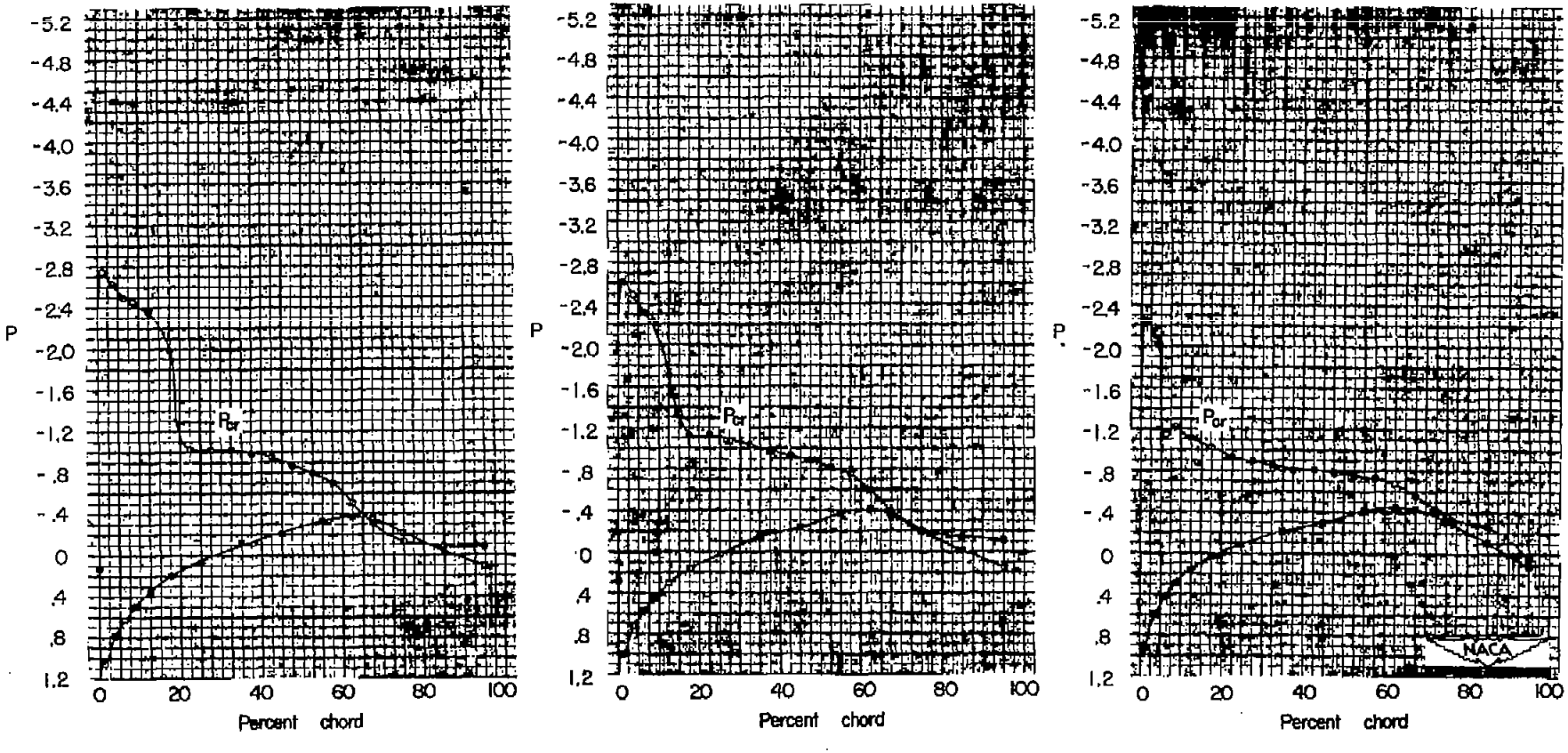
Figure 11.- Continued. $M = 0.60$.



Station 1 Station 2 Station 3

(h) $\alpha = 11.18^{\circ}$.

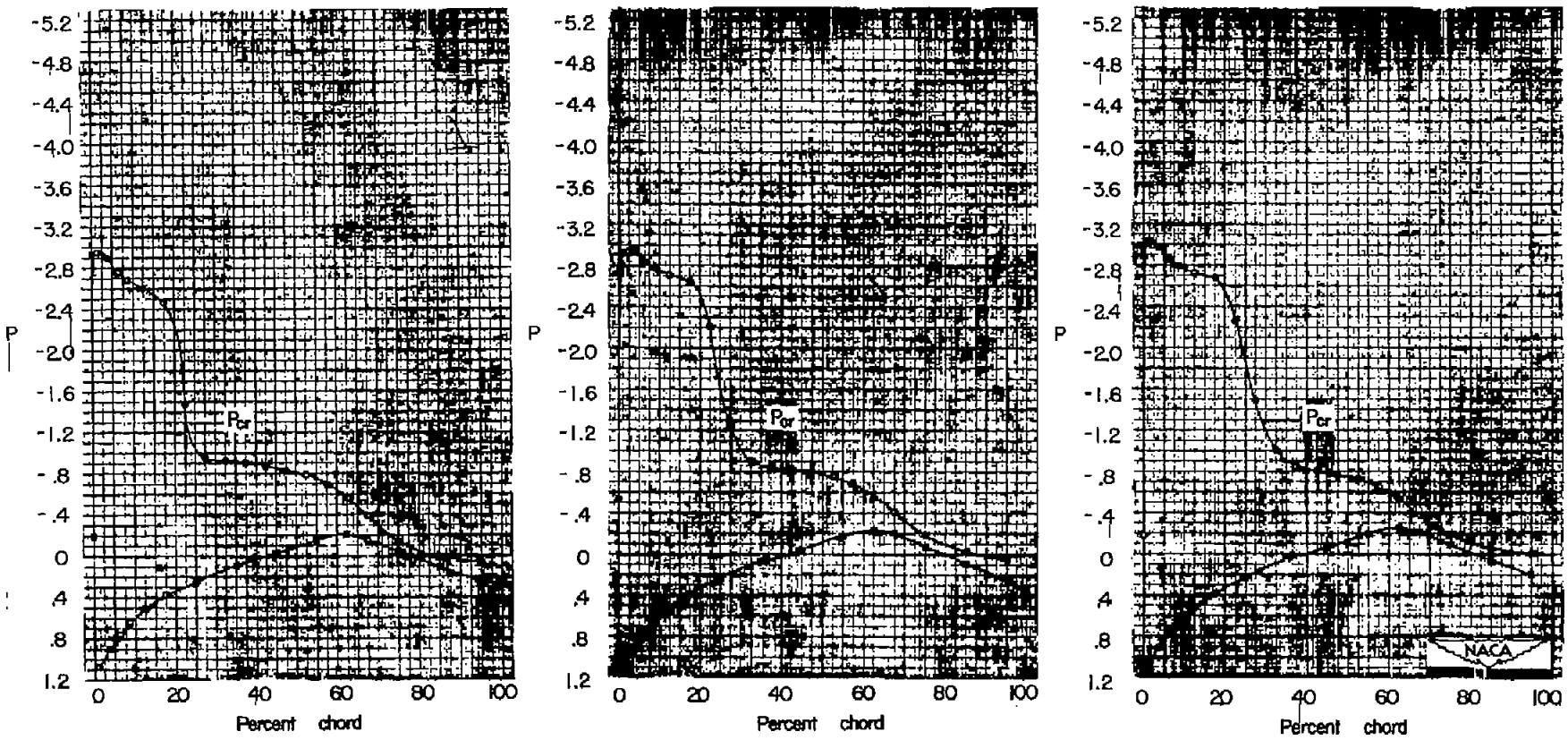
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(h) Concluded. $\alpha = 11.18^{\circ}$.

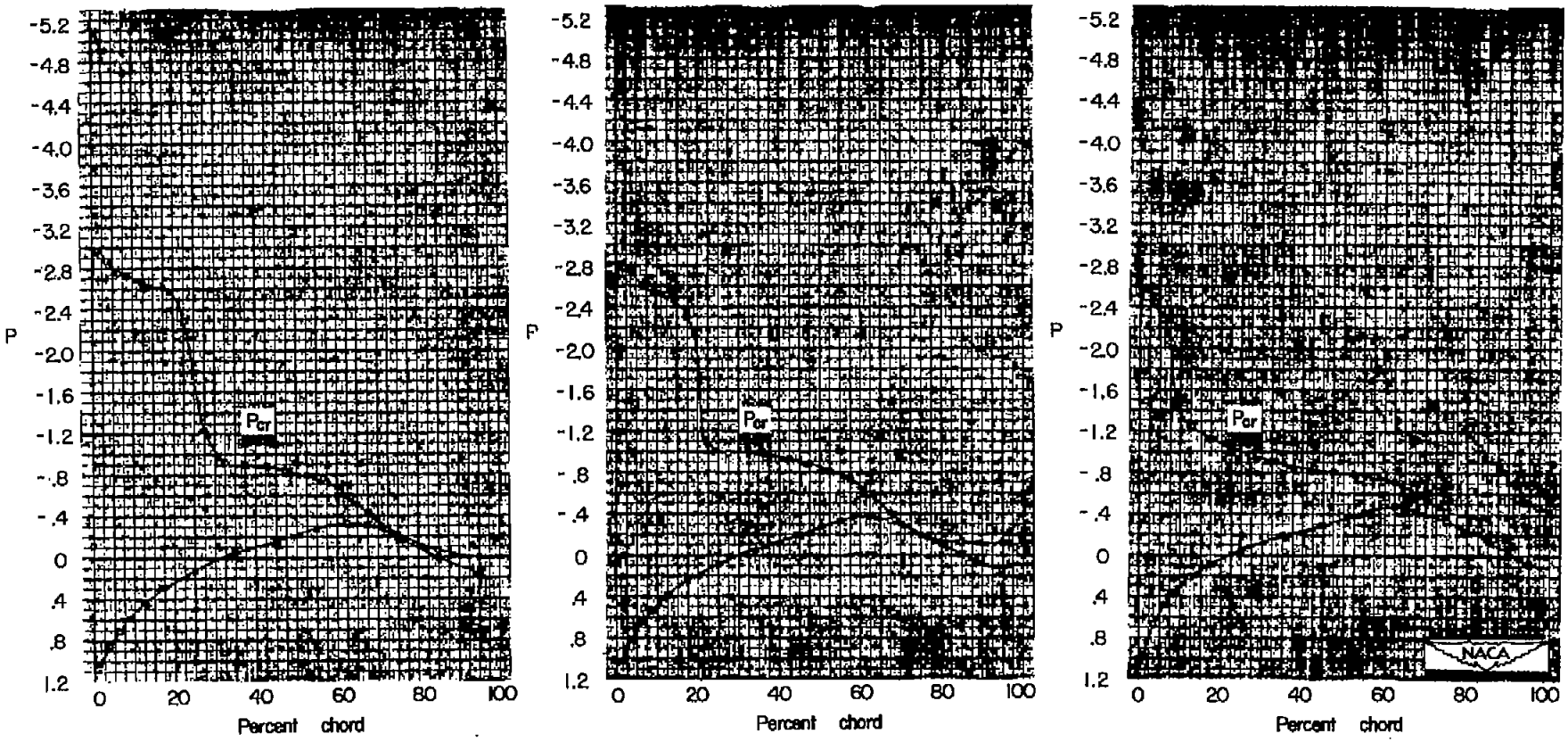
Figure 11.- Continued. M = 0.60.



Station 1 Station 2 Station 3

(i) $\alpha = 12.82^{\circ}$.

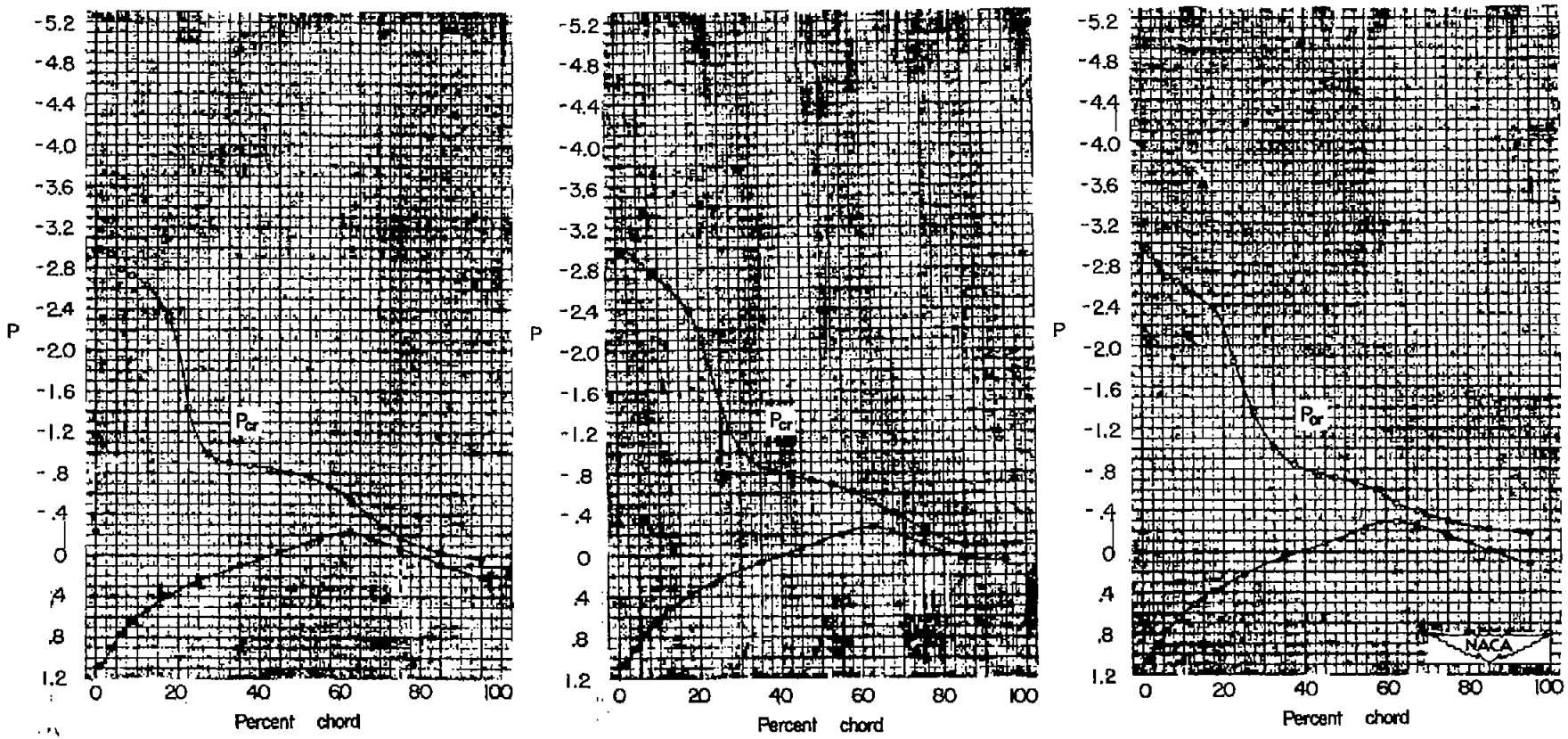
Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(i) Concluded. $\alpha = 12.82^{\circ}$.

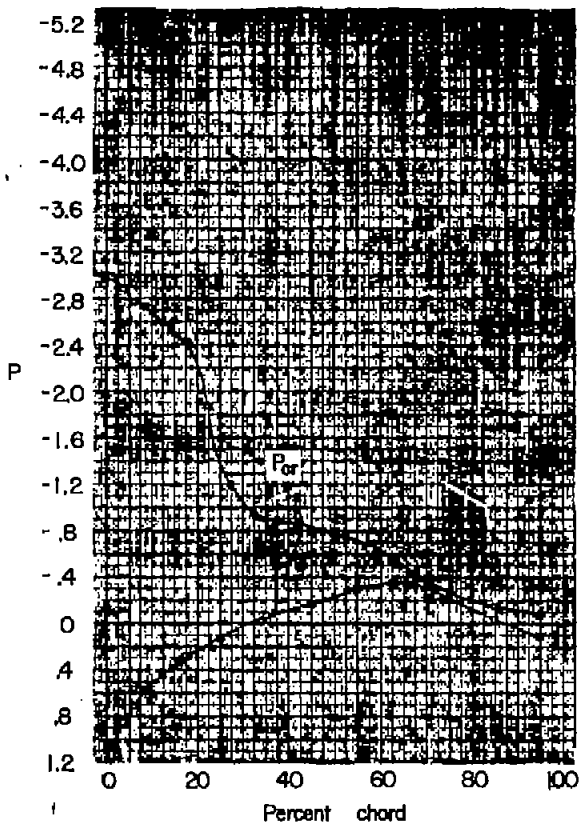
Figure 11.- Continued. M = 0.60.



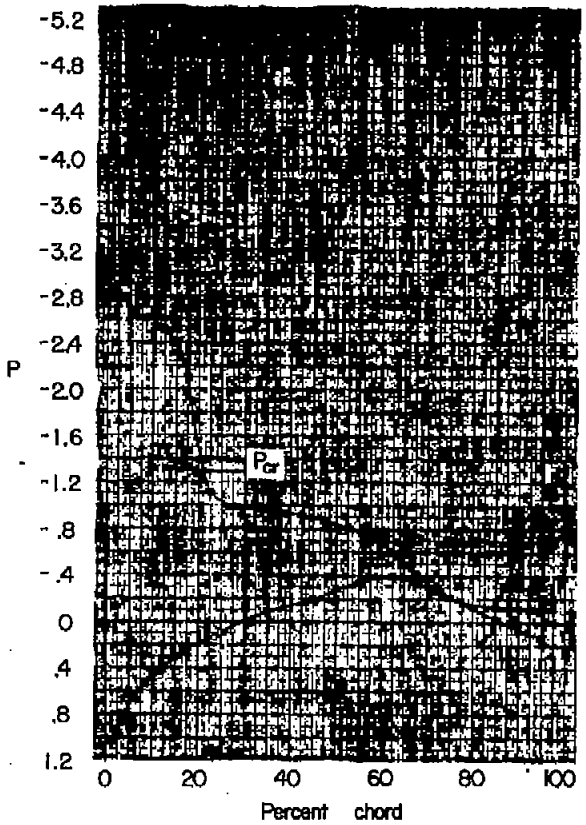
Station 1 Station 2 Station 3

(j) $\alpha = 13.31^{\circ}$.

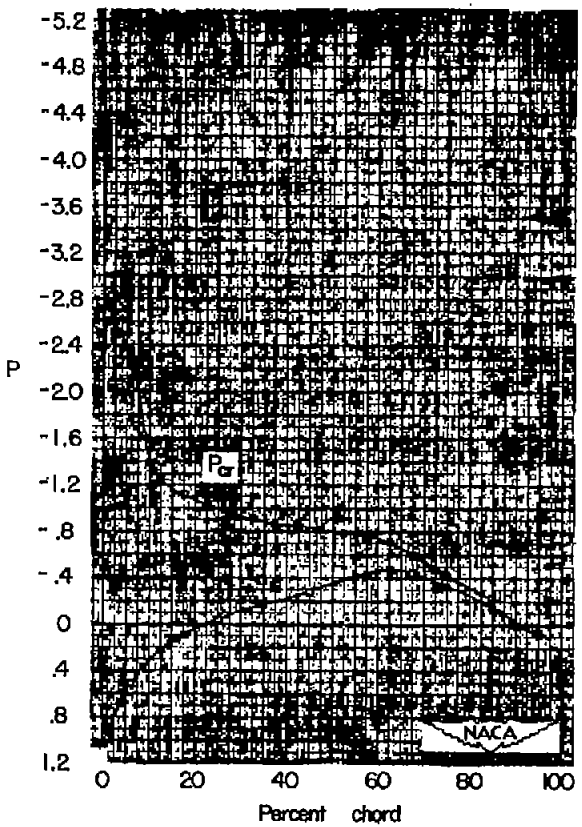
Figure 11.- Continued. M = 0.60.



Station 4



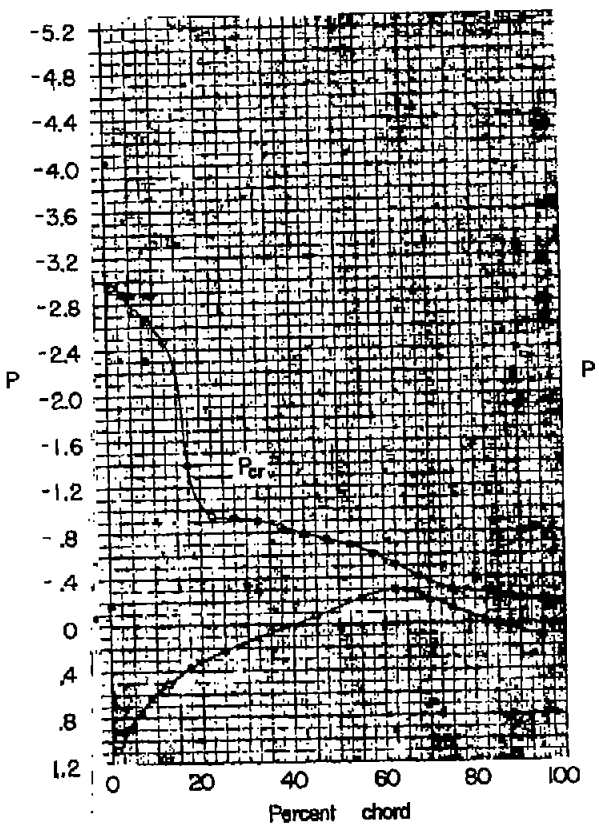
Station 5



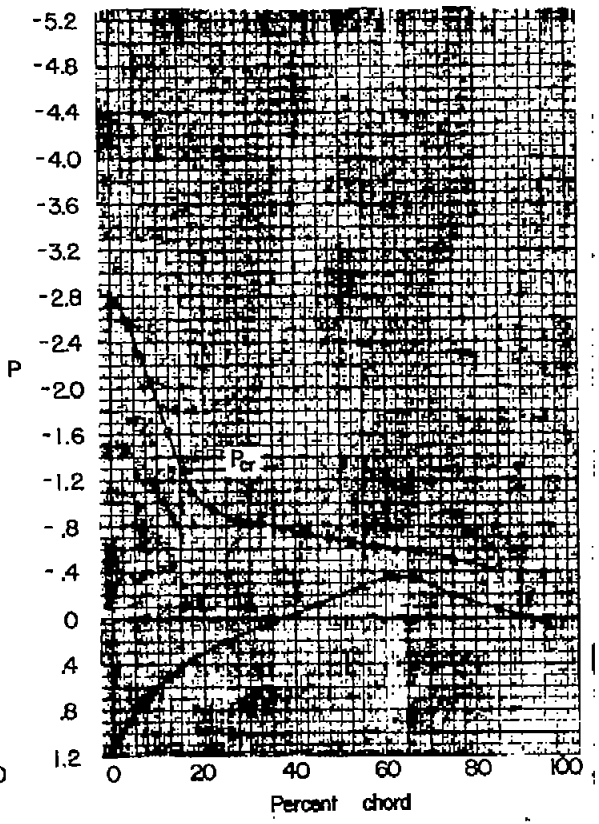
Station 6

(j) Concluded. $\alpha = 13.31^{\circ}$.

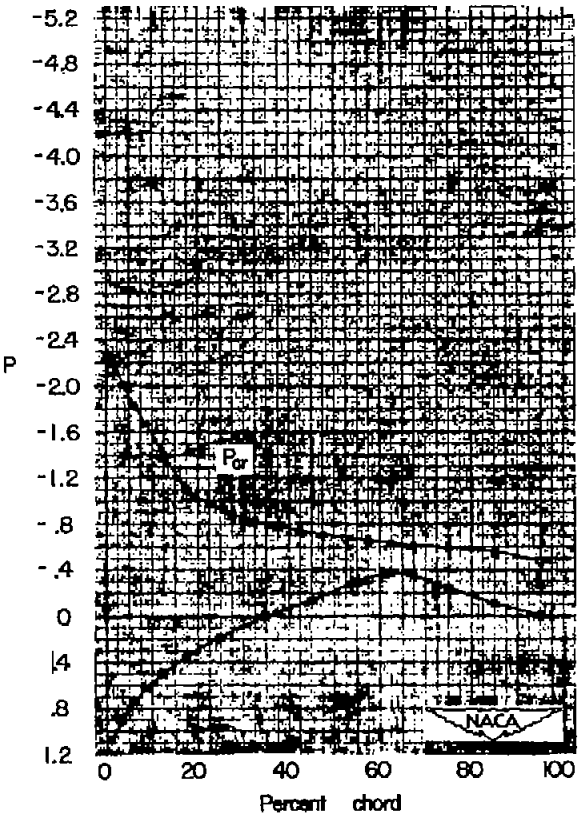
Figure 11.- Continued. M = 0.60.



Station 1



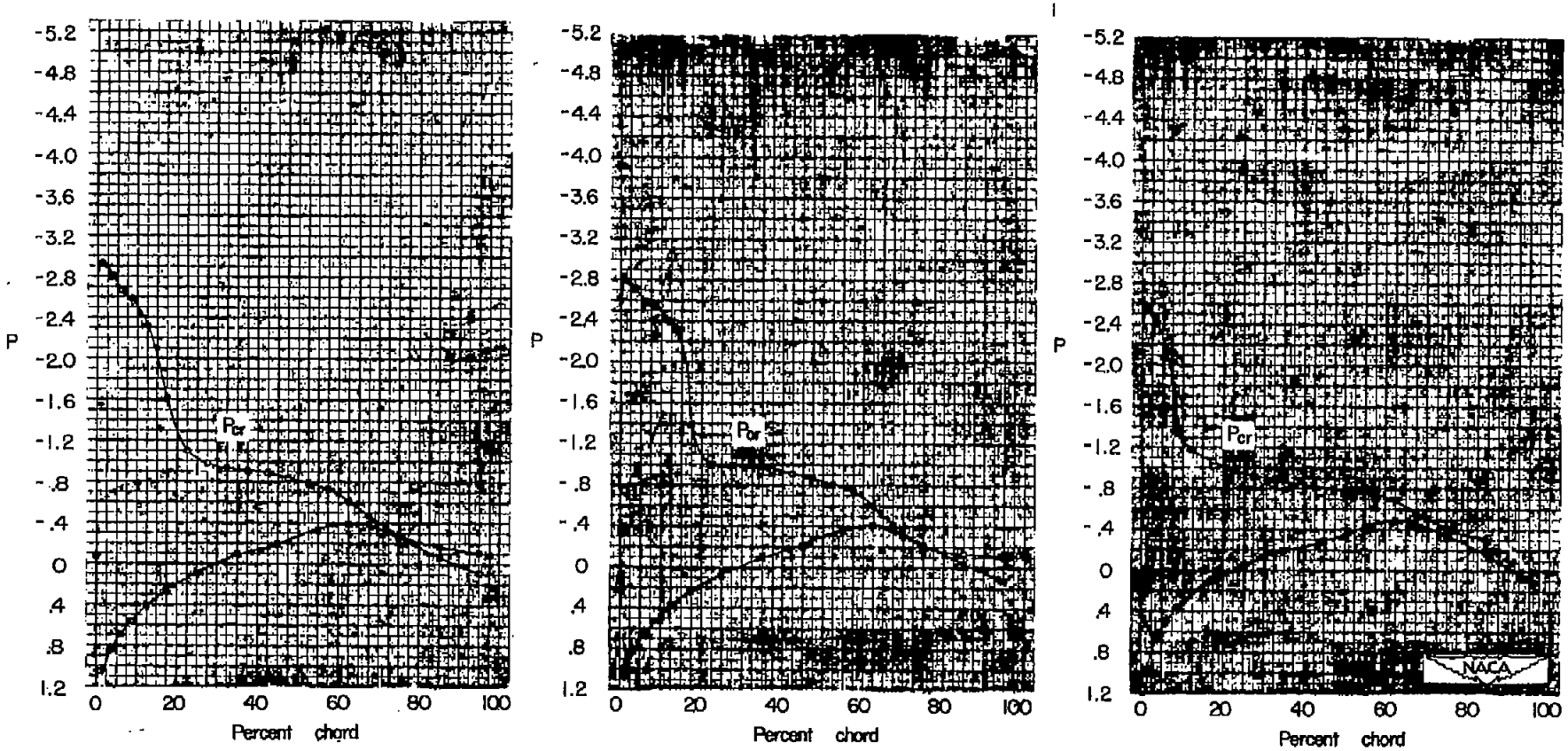
Station 2



Station 3

(k) $\alpha = 13.60^{\circ}$.

Figure 11.- Continued. M = 0.60.



Station 4 Station 5 Station 6

(k) Concluded. $\alpha = 13.60^{\circ}$.

Figure 11.- Concluded. M = 0.60.