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MODEL AND BOUNDARY AERODYNAMIC DATA FROM
HIGH BLOCKAGE TWO-DIMENSIONAL AIRFOIL TESTS
IN A SHALLOW UNSTREAMLINED TRANSONIC
FLEXIBLE WALLED TEST SECTION

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CONTENTS

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1. TRANSONIC SELF-STREAMLINING WIND TUNNEL DATA

In the course of previously reported two-dimensional tests on a 4 inch (10.16cm) chord NACA 0012-64 airfoil in the Transonic Self-Streamlining Wind Tunnel (TSWT)[1,2], twenty-four runs were performed with the flexible floor and ceiling of the test section set 'straight'. These runs provide information on the gross boundary interference present in a small non-porous test section (with a nominally 6 inches {15.24cm} square cross-section), where the model blockage is 8% at $\alpha=0$. These tests are discussed here because the associated data may prove useful in the development of wind tunnel correction techniques.

Four sets of 'straight wall' contours have been obtained experimentally which give constant wall Mach numbers of 0.3, 0.5, 0.7 or 0.9 in an empty test section. The walls in fact diverge to allow for boundary layer growth. However, the walls are effectively straight in the aerodynamic sense with no model present. With a model installed, the wall boundary layer displacement thickness is altered by the introduction of longitudinal pressure gradients. The walls are then no longer effectively 'straight'. Wall adjustments to correct the contours have not been attempted in this work. This point is raised as a warning if the 'straight wall' velocity distributions are to be used for any form of wind tunnel corrections.

The 'straight wall' runs are summarised in Table 1. Assessment of wall induced effects on the flow at the model have been made using the top and bottom wall loadings. These wall loadings are determined from the imbalance between real measured wall pressures (inside the test section) and imaginary calculated wall pressures (external to the test section). The wall induced effects are referred to as 'Residual Interferences' and have been calculated in terms of induced angle of attack ($\Delta\alpha$), induced camber (assessed as an effect on C_L and tabulated as ΔC_L) and induced Mach number perturbation (ΔM). The force coefficients C_L and C_D were determined from integrated model pressures. Notice that for each run with zero angle of attack, the bottom wall supports a larger

E (average of the modulus of the pressure coefficient error between real and imaginary flows along a flexible wall) than the top wall. Since the imaginary flowfields above and below the test section are uniform and undisturbed for 'straight wall' cases, the difference between top and bottom wall E is attributed to the asymmetry of model position between the flexible walls. This may be due to some test section centreline displacement or curvature introduced during the experimental determination of 'straight wall' contours. The high values of E for the straight wall runs imply high levels of interference induced by each wall at the model.

The 'straight wall'model pressure distributions are plotted on Figure 1 and tabulated in Table 2. Corresponding Mach number distributions along the flexible walls are shown in Figure 2. It is evident from the wall data that the peak wall Mach numbers rise rapidly with increasing freestream Mach number (for example, compare wall data for the $\alpha=4^{o}$, runs 66, 40 & 42). This effect of flow compressibility leads to choking of the test section which, of course, sets an upper limit to the Mach number range of 'straight wall' tests.

The 'straight wall' model data (α, C_L and C_D) has been corrected for interference induced by a non-porous test section boundary, using the conventional technique developed by Allen and Vincenti[3]. Also, the corrections due to residual interferences have been applied to the model data (α and C_L). Several options exist regarding the application of the corrections for these residual interferences. Corrections can be applied independently to angle of attack, lift and Mach number. Alternatively, streamwise lift can be corrected for induced camber and Mach number perturbations while angle of attack is corrected independently. Finally, the three components of the residual interferences can be related to corrections to lift while α and M remain constant. To assist with data comparisons, only the lift and angle of attack have been corrected for residual interferences, with no correction applied to the freestream Mach number.

Wall-induced errors at the model are assessed from the wall loadings made incompressible by use of linearised theory. In TSWT operation the residual interferences are made small by wall contouring to streamline shapes, even up to freestream Mach numbers of 0.85. In general, only small model corrections can be applied with confidence at transonic speeds. Hence, in these circumstances, simple incompressible assessment of residual interferences can be used over a wide range of test Mach number.

The corrected model data is summarised in Figure 3 where the lift curve slope ($dC_L/\ell\alpha$) is plotted against freestream Mach number. Results from TSWT straight wall and streamlined wall[2] tests are shown together with theoretical curves derived from linearised theory which are constrained to pass through the lowest Mach number data point on each data set. There is reasonable agreement between theory and experiment, especially for the streamlined wall case. The model data corrected for residual errors is very encouraging considering the gross boundary interference present in the 'straight wall' tests. The Allen and Vincenti corrections appear too large, particularly at the higher Mach numbers, illustrating the inaccuracy of applying only simple corrections to the overall model forces which take no account of detailed changes in the model flow pattern - for example, changes in model shock positions.

The straight wall data was obtained at freestream Mach numbers of approximately 0.7, 0.5 and 0.3. 'Straight wall' testing above Mach No.0·7 was impractical with this model. The walls were set to the appropriate 'straight wall' contours which corresponded closely to the freestream Mach number of the test. The variation of C_L and C_D with α for the three Mach numbers are shown on Figures 4 and 5 respectively together with streamlined wall data and corrected data where appropriate.

The C_L data can be conveniently summarised by the fitting of a least squares curve to each set of data over the range $-8^o<\alpha<+8^o$ The straight line slopes and zero α intercepts are:-

Data	Slope $dC_L/d\alpha$ per degree			Zero α intercept C_L		
Mach No.	0.7	0.5	0.3	0.7	0.5	0.3
Straight wall data	.1563	.1197	.1091	-.0898	-.0752	-.0602
Straight wall data corrected by Allen & Vincenti method.	.0927	.0875	.0842	-.0519	-.0552	-.0511
Straight wall data corrected for residual interferences	.1203	.0916	.0895	-.0775	-.0672	-.524
Streamlined wall data[2]	.1178	.0965	-	-.0753	-.0574	-

If the streamlined data is assumed correct as suggested, by comparison with reference data, then the 'straight wall' residual corrections seem very good. The ratio of lift curve slopes for streamlined wall data and residual corrected straight wall data is .97 and 1.05 for Mach numbers 0.7 and 0.5 respectively.

The C_D data as shown on Figure 5 relates only to pressure drag. While magnitudes are aerodynamically meaningless, the symmetry of the C_D curves about the zero α axis is shown. Future work will investigate the momentum defect in the airfoil wake, to assess model drag.

The 'straight wall' data is presented here in graphical and tabulated form to conclude the summary of current TSWT tests with the NACA 0012-64 airfoil section. Because of the high blockage, this data may prove useful to those engaged in the development of interference correction methods for transonic wind tunnel testing.

2. LIST OF SYMBOLS

α	Angle of attack
c	Model chord
C_C	Chordwise force coefficient
C_L	Lift coefficient
C_D	Pressure drag coefficient
C_M	Pitching moment about the leading edge
C_N	Normal force coefficient
C_p	Pressure coefficient
E	Average of the modulus of the pressure coefficient error between real and imaginary flows along a flexible wall $\left(\dfrac{\sum_{1}^{n} \lvert C_{p_r} - C_{p_i} \rvert}{n}\right)$
M_∞	Freestream Mach number
n	Number of jacks along a wall
R_c	Chord's Reynolds number
X	Distance from leading edge

SUFFIXES

'~'	Uncorrected data
'i'	imaginary
'r'	real
'TOP'	Top wall
'BOT'	Bottom wall

3. REFERENCES

1. M.J. Goodyer and S.W.D. Wolf — 'The Development of a Self-Streamlining Flexible Walled Transonic Test Section' AIAA Paper 80-0440, March 1980

2. S.W.D. Wolf — 'Selected Data from a Transonic Flexible Walled Test Section' NASA CR-159360, September 1980

3. H.J. Allen and W.G. Vincenti — 'Wall Interference in a Two-Dimensional Flow Wind Tunnel with Consideration of the Effect of Compressibility' NACA Report 782, 1944

TABLE 1 SUMMARY OF TSWT 'STRAIGHT WALL' DATA

Fig. No. Suffix	Run No.	Model α	Mach No.	Model Data		Residual Interferences			E_{TOP}	E_{BOT}
				C_L'	C_D'	Δα	ΔC_L	ΔM		
1	66	4°	.706	.5466	.032	+.5	.0649	.041	.1318	.0665
2	56	3°	.697	.3854	.0027	+.26	.0557	.029	.0897	.0431
3	55	2°	.693	.2352	-.004	+.15	.033	.025	.069	.0417
4	54	0°	.683	-.1111	-.0109	-.115	-.0684	.023	.042	.0573
5	68	-2°	.701	-.4636	.0013	-.413	-.0491	.033	.0462	.1031
6	67	-4°	.701	-.6624	.0505	-.654	-.0534	.051	.089	.1742
7	40	4°	.520	.4089	-.003	.255	.0629	.015	.0751	.0236
8	53	3°	.505	.2697	-.006	.302	.0692	.011	.0665	.0194
9	39	2°	.516	.1755	-.0098	.863	.0288	.013	.0499	.0271
10	36	0°	.505	-.0728	-.0136	-.097	-.0057	.012	.0290	.0406
11	52	-2°	.499	-.3195	-.0136	-.222	-.0424	.013	.0195	.0609
12	51	-3°	.505	-.4415	-.0124	-.298	-.0591	.014	.018	.0724
13	50	-4°	.504	-.5467	-.0092	-.363	-.0742	.015	.0182	.0857
14	44	10°	.301	.9753	.0565	.719	.1432	.013	.1485	.0473
15	43	8°	.298	.8317	.0363	.637	.1186	.011	.1253	.0385
16	42A	6°	.299	.5872	.0133	.411	.0877	.009	.093	.024
17	42	4°	.304	.3658	-.0058	.221	.0583	.008	.0654	.0193
18	41	2°	.296	.1608	-.0109	.103	.0237	.007	.045	.0217
19	40A	0°	.293	-.0695	-.0131	-.067	-.0094	.006	.0265	.0385
20	45	-2°	.297	-.2801	-.0119	-.207	-.0394	.007	.0153	.0573
21	46	-4°	.296	-.4871	-.0052	-.4	-.0631	.008	.0181	.0856
22	47	-6°	.300	-.7399	.0095	-.53	-.1012	.009	.0338	.1078
23	48	-8°	.296	-.9106	.0261	-.563	-.1301	.013	.0378	.1396
24	49	-10°	.301	-1.052	.0517	-.567	-.1509	.015	.045	.1688

TABLE 2

NACA 0012-64

PRESSURE DISTRIBUTIONS

AND FORCES.

TABLE 2.1

NACA SECTION ANALYSIS
0012-64

RUN NO. = 66

ALPHA = [illegible]

MACH NO. =0.7058

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 19

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0116	-0.0116
1	-0.6274	0.6042
2	-0.9894	0.3056
.5	-1.2049	0.0949
7	-1.2337	-0.0363
9	-1.2423	-0.1105
15	-1.2114	-0.2003
20	-1.2715	-0.2624
25	-1.2749	-0.3228
29	-1.2822	-0.3557
35	-1.2805	-0.4056
40	-1.2323	-0.4416
44	-1.2444	-0.4812
50	-1.2099	-0.4966
55	-1.2274	-0.5163
60	-1.1239	-0.5129
64	-0.7273	-0.5020
70	-0.4999	-0.4769
75	-0.3835	-0.4435
80	-0.2949	-0.4142
85	-0.2085	-0.3633
90	-0.1150	-0.2175
95	-0.0223	-0.1353

	UPPER	LOWER	TOTAL
CN	0.8644	-0.3169	0.5475
CC	-0.0314	0.0251	-0.0063
CM	-0.3214	0.1796	-0.1418

AIRFOIL PERFORMANCE

CL	CD	CM
0.5466	0.0320	-0.1418

TABLE 2.2

NACA SECTION ANALYSIS
0012-64

RUN NO. = 56

ALPHA = 3.0

MACH NO. =0.697

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 10

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0163	-0.0163
1	-0.5707	0.5382
2	-0.9447	0.2357
5	-1.1186	0.0352
7	-1.1501	-0.0881
9	-1.1677	-0.1532
15	-1.1414	-0.2237
20	-1.1519	-0.2783
25	-1.0677	-0.3276
29	-0.9960	-0.3442
35	-0.9348	-0.3881
40	-0.7947	-0.4091
44	-0.7422	-0.4337
50	-0.7002	-0.4372
55	-0.6560	-0.4414
60	-0.6015	-0.4309
64	-0.5192	-0.3999
70	-0.4482	-0.3750
75	-0.3658	-0.3495
80	-0.2685	-0.3026
85	-0.1648	-0.1904
90	-0.0547	-0.0987
95	0.0547	-0.0135

	UPPER	LOWER	TOTAL
CN	0.6534	-0.2684	0.3850
CC	-0.0343	0.0167	-0.0175
CM	-0.2239	0.1391	-0.0848

AIRFOIL PERFORMANCE

CL	CD	CM
0.3854	0.0027	-0.0848

TABLE 2.3

NACA SECTION ANALYSIS
0012-64

RUN NO. = 55

ALPHA = 2.0

MACH NO. =0.6927

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 14

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	0.0059	0.0059
1	-0.3569	0.3687
2	-0.7156	0.0567
5	-0.8487	-0.1170
7	-0.8956	-0.2191
9	-0.8761	-0.2775
15	-0.8089	-0.3234
20	-0.7859	-0.3693
25	-0.7576	-0.4029
29	-0.7493	-0.4184
35	-0.7405	-0.4485
40	-0.7104	-0.4538
44	-0.7069	-0.4716
50	-0.6821	-0.4680
55	-0.6417	-0.4581
60	-0.5744	-0.4386
64	-0.4891	-0.4072
70	-0.4180	-0.3835
75	-0.3469	-0.3455
80	-0.2672	-0.2298
85	-0.1767	-0.1501
90	-0.0690	-0.0646
95	0.0517	0.0345

	UPPER	LOWER	TOTAL
CN	0.5383	-0.3034	0.2349
CC	-0.0210	0.0088	-0.0122
CM	-0.1987	0.1404	-0.0584

AIRFOIL PERFORMANCE

CL	CD	CM
0.2352	-0.0040	-0.0584

TABLE 2.4

NACA SECTION ANALYSIS
0012-64

RUN NO. = 54

ALPHA = 0.0

MACH NO. =0.6831

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 15

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	0.0077	0.0077
1	0.1891	-0.1737
2	-0.1207	-0.4977
5	-0.2954	-0.5900
7	-0.3765	-0.6309
9	-0.4000	-0.6671
15	-0.4307	-0.6417
20	-0.4488	-0.6526
25	-0.4615	-0.6580
29	-0.4745	-0.6350
35	-0.4907	-0.6404
40	-0.4781	-0.6224
44	-0.4925	-0.6206
50	-0.4889	-0.5953
55	-0.4760	-0.5747
60	-0.4561	-0.5493
64	-0.4182	-0.4736
70	-0.3927	-0.3796
75	-0.3286	-0.3177
80	-0.2266	-0.2368
85	-0.1515	-0.1588
90	-0.0648	-0.0517
95	0.0386	0.0634

	UPPER	LOWER	TOTAL
CN	0.3439	-0.4550	-0.1111
CC	0.0013	-0.0122	-0.0109
CM	-0.1475	0.1753	0.0279

AIRFOIL PERFORMANCE

CL	CD	CM
-0.1111	-0.0109	0.0279

TABLE 2.5

NACA SECTION ANALYSIS
0012-64

RUN NO. = 68

ALPHA = -2.0

MACH NO. =0.7008

WING DATA FILE NAME = *WING2.DAT
INPUT FILE NO. - 3

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0458	-0.0458
1	0.5164	-0.6080
2	0.2198	-0.9389
5	-0.0035	-1.1654
7	-0.1044	-1.0958
9	-0.1618	-1.0958
15	-0.2418	-1.0940
20	-0.2818	-1.1514
25	-0.3183	-1.1375
29	-0.3560	-1.1149
35	-0.3803	-1.1115
40	-0.4064	-1.0906
44	-0.4376	-1.1062
50	-0.4585	-1.0594
55	-0.4634	-0.9233
60	-0.4460	-0.6080
64	-0.4249	-0.5364
70	-0.4016	-0.4785
75	-0.3712	-0.4100
80	-0.3345	-0.3091
85	-0.2265	-0.2160
90	-0.1080	-0.0917
95	-0.0106	0.0254

	UPPER	LOWER	TOTAL
CN	0.2819	-0.7452	-0.4633
CC	0.0162	-0.0311	-0.0149
CM	-0.1463	0.2707	0.1243

AIRFOIL PERFORMANCE

CL	CD	CM
-0.4636	0.0013	0.1243

TABLE 2.6

NACA SECTION ANALYSIS
0012-64

RUN NO. = 67

ALPHA = -4.0

MACH NO. =0.701

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 20

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0930	-0.0930
1	0.6434	-0.8295
2	0.3625	-1.1100
5	0.1179	-1.3611
7	0.0000	-1.3521
9	-0.0728	-1.3538
15	-0.1786	-1.3158
20	-0.2359	-1.3625
25	-0.2931	-1.3763
29	-0.3509	-1.4087
35	-0.4047	-1.3861
40	-0.4464	-1.3288
44	-0.5020	-1.4000
50	-0.5593	-1.4000
55	-0.5749	-1.3838
60	-0.5870	-1.3245
64.	-0.5942	-1.0657
70	-0.5809	-0.8328
75	-0.5507	-0.6405
80	-0.5178	-0.4792
85	-0.4785	-0.3690
90	-0.4097	-0.2371
95	-0.2427	-0.1593

	UPPER	LOWER	TOTAL
CN	0.3611	-1.0254	-0.6643
CC	0.0319	-0.0278	0.0041
CM	-0.2175	0.4091	0.1916

AIRFOIL PERFORMANCE

CL	CD	CM
-0.6624	0.0505	0.1916

TABLE 2.7

NACA SECTION ANALYSIS
0012-64

RUN NO. = 40

ALPHA = 4.0

MACH NO. =0.5203

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 8

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.3297	-0.3297
1	-1.3940	0.7345
2	-1.4600	0.4416
5	-1.2476	0.2281
7	-1.1235	0.0984
9	-1.0240	0.0324
15	-0.8608	-0.0514
20	-0.7512	-0.1017
25	-0.7110	-0.1476
29	-0.6775	-0.1699
35	-0.6484	-0.2023
40	-0.6193	-0.2258
44	-0.5891	-0.2415
50	-0.5567	-0.2471
55	-0.5042	-0.2504
60	-0.4651	-0.2515
64	-0.4114	-0.2415
70	-0.3466	-0.2225
75	-0.2828	-0.2023
80	-0.2046	-0.1777
85	-0.1185	-0.1498
90	-0.0257	-0.0682
95	0.0682	0.0067

	UPPER	LOWER	TOTAL
CN	0.5298	-0.1221	0.4077
CC	-0.0495	0.0180	-0.0315
CM	-0.1683	0.0779	-0.0904

AIRFOIL PERFORMANCE

CL	CD	CM
0.4089	-0.0030	-0.0904

TABLE 2.8

NACA SECTION ANALYSIS
0012-64

RUN NO. = 53

ALPHA = 3.0

MACH NO. =0.505

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 9

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.1623	-0.1623
1	-0.8736	0.5489
2	-1.0161	0.2476
5	-0.9156	0.0654
7	-0.8152	-0.0409
9	-0.7720	-0.0923
15	-0.6984	-0.1542
20	-0.6353	-0.1892
25	-0.5980	-0.2219
29	-0.5711	-0.2348
35	-0.5478	-0.2593
40	-0.5326	-0.2756
44	-0.5092	-0.2826
50	-0.4894	-0.2850
55	-0.4473	-0.2791
60	-0.4181	-0.2745
64	-0.3749	-0.2604
70	-0.3212	-0.2383
75	-0.2628	-0.2149
80	-0.1950	-0.1880
85	-0.1156	-0.1261
90	-0.0234	-0.0420
95	0.0759	0.0200

	UPPER	LOWER	TOTAL
CN	0.4366	-0.1676	0.2690
CC	-0.0327	0.0123	-0.0204
CM	-0.1466	0.0874	-0.0593

AIRFOIL PERFORMANCE

CL	CD	CM
0.2697	-0.0063	-0.0593

TABLE 2.9

NACA SECTION ANALYSIS
0012-64

RUN NO. = 39

ALPHA = 2.0

MACH NO. =0.516

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 9

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0820	-0.0820
1	-0.5249	0.3609
2	-0.7240	0.0645
5	-0.7138	-0.0803
7	-0.6901	-0.1629
9	-0.6482	-0.2059
15	-0.6018	-0.2455
20	-0.5577	-0.2681
25	-0.5328	-0.2930
29	-0.5159	-0.2998
35	-0.4989	-0.3156
40	-0.4876	-0.3269
44	-0.4740	-0.3315
50	-0.4582	-0.3247
55	-0.4208	-0.3145
60	-0.3960	-0.3043
64	-0.3552	-0.2851
70	-0.3077	-0.2613
75	-0.2557	-0.2342
80	-0.1957	-0.1742
85	-0.1233	-0.1063
90	-0.0351	-0.0339
95	0.0690	0.0430

	UPPER	LOWER	TOTAL
CN	0.3876	-0.2126	0.1751
CC	-0.0221	0.0062	-0.0159
CM	-0.1373	0.0973	-0.0399

AIRFOIL PERFORMANCE

CL	CD	CM
0.1755	-0.0098	-0.0399

TABLE 2.10

NACA SECTION ANALYSIS
0012-64

RUN NO. = 36

ALPHA = 0.0

MACH NO. =0.505

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 10

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0418	-0.0418
1	0.1371	-0.2207
2	-0.1429	-0.4635
5	-0.2683	-0.4800
7	-0.3137	-0.4879
9	-0.3253	-0.4972
15	-0.3415	-0.4763
20	-0.3392	-0.4635
25	-0.3404	-0.4612
29	-0.3497	-0.4472
35	-0.3555	-0.4461
40	-0.3590	-0.4438
44	-0.3578	-0.4333
50	-0.3543	-0.4136
55	-0.3357	-0.3915
60	-0.3229	-0.3683
64	-0.3032	-0.3299
70	-0.2776	-0.2835
75	-0.2300	-0.2323
80	-0.1673	-0.1743
85	-0.1069	-0.1115
90	-0.0325	-0.0256
95	0.0592	0.0709

	UPPER	LOWER	TOTAL
CN	0.2539	-0.3267	-0.0728
CC	-0.0014	-0.0122	-0.0136
CM	-0.1049	0.1223	0.0174

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0728	-0.0136	0.0174

TABLE 2.11

NACA SECTION ANALYSIS
0012-64

RUN NO. = 52

ALPHA = -2.0

MACH NO. =0.499

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 11

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.2193	-0.2193
1	0.5773	-1.0160
2	0.2785	-1.1308
5	0.1380	-0.9730
7	-0.0024	-0.8654
9	-0.0598	-0.8308
15	-0.1291	-0.7327
20	-0.1554	-0.6706
25	-0.1829	-0.6383
29	-0.2056	-0.5953
35	-0.2271	-0.5785
40	-0.2438	-0.5630
44	-0.2510	-0.5355
50	-0.2618	-0.4997
55	-0.2510	-0.4662
60	-0.2510	-0.4315
64	-0.2355	-0.3801
70	-0.2164	-0.3239
75	-0.1901	-0.2654
80	-0.1554	-0.1901
85	-0.1052	-0.1148
90	-0.0394	-0.0155
95	0.0430	0.0777

	UPPER	LOWER	TOTAL
CN	0.1405	-0.4594	-0.3189
CC	0.0122	-0.0369	-0.0247
CM	-0.0755	0.1518	0.0762

AIRFOIL PERFORMANCE

CL	CD	CM
-0.3195	-0.0136	0.0762

TABLE 2.12

NACA SECTION ANALYSIS
0012-64

RUN NO. = 51

ALPHA = -3.0

MACH NO. =0.5047

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 12

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.3850	-0.3850
1	0.7382	-1.5082
2.	0.4474	-1.5495
5	0.2849	-1.2798
7	0.1224	-1.0997
9	0.0506	-1.0149
15	-0.0389	-0.8430
20	-0.0789	-0.7794
25	-0.1142	-0.7453
29	-0.1460	-0.6970
35	-0.1731	-0.6652
40	-0.1954	-0.6382
44	-0.2108	-0.5993
50	-0.2249	-0.5510
55	-0.2202	-0.5098
60	-0.2249	-0.4663
64	-0.2155	-0.4097
70	-0.2002	-0.3438
75	-0.1778	-0.2779
80	-0.1495	-0.1978
85	-0.1083	-0.1142
90	-0.0495	-0.0141
95	0.0247	0.0718

	UPPER	LOWER	TOTAL
CN	0.0976	-0.5378	-0.4402
CC	0.0169	-0.0523	-0.0355
CM	-0.0654	0.1693	0.1039

AIRFOIL PERFORMANCE

CL	CD	CM
-0.4415	-0.0124	0.1039

TABLE 2.13

NACA SECTION ANALYSIS
0012-64

RUN NO. = 50

ALPHA = -4.0

MACH NO. =0.5043

WING DATA FILE NAME = %STWD.DAT
INPUT FILE NO. - 13

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.5778	-0.5778
1	0.8512	-2.0069
2	0.5784	-1.9226
5	0.4010	-1.6170
7	0.2236	-1.2517
9	0.1428	-1.1123
15	0.0386	-0.9730
20	-0.0117	-0.8875
25	-0.0550	-0.8337
29	-0.0937	-0.7716
35	-0.1241	-0.7306
40	-0.1510	-0.6955
44	-0.1686	-0.6487
50	-0.1885	-0.5925
55	-0.1885	-0.5421
60	-0.1979	-0.4929
64	-0.1920	-0.4262
70	-0.1838	-0.3571
75	-0.1663	-0.2834
80	-0.1428	-0.1979
85	-0.1077	-0.1136
90	-0.0574	-0.0187
95	0.0070	0.0562

	UPPER	LOWER	TOTAL
CN	0.0598	-0.6046	-0.5448
CC	0.0198	-0.0672	-0.0473
CM	-0.0556	0.1836	0.1279

AIRFOIL PERFORMANCE

CL	CD	CM
-0.5467	-0.0092	0.1279

TABLE 2.14

NACA SECTION ANALYSIS
0012-64

RUN NO. = 44

ALPHA = 10.0

MACH NO. =0.3011

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 3

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-1.6738	-1.6738
1	-4.3475	0.9990
2	-4.2293	0.9842
5	-2.4619	0.7980
7	-2.0895	0.6295
9	-1.8058	0.5409
15	-1.4659	0.3872
20	-1.2531	0.2926
25	-1.1172	0.2187
29	-1.0137	0.1626
35	-0.9221	0.1064
40	-0.8423	0.0591
44	-0.7714	0.0207
50	-0.6975	-0.0089
55	-0.6088	-0.0384
60	-0.5379	-0.0650
64	-0.4581	-0.0828
70	-0.3665	-0.0857
75	-0.2808	-0.0975
80	-0.2010	-0.0946
85	-0.1360	-0.0946
90	-0.0946	-0.0798
95	-0.0650	-0.0709

	UPPER	LOWER	TOTAL
CN	0.8656	0.1047	0.9703
CC	-0.1375	0.0238	-0.1137
CM	-0.2318	0.0068	-0.2250

AIRFOIL PERFORMANCE

CL	CD	CM
0.9753	0.0565	-0.2250

TABLE 2.15

NACA SECTION ANALYSIS
0012-64

RUN NO. = 43

ALPHA = 8.0

MACH NO. =0.298

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 4

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-1.1472	-1.1472
1	-3.3088	1.0144
2	-2.9586	0.8996
5.	-1.9955	0.6732
7	-1.7117	0.5042
9	-1.5125	0.4166
15	-1.2559	0.2808
20	-1.0868	0.2023
25	-0.9812	0.1328
29	-0.9027	0.0906
35	-0.8393	0.0423
40	-0.7759	0.0060
44	-0.7215	-0.0211
50	-0.6642	-0.0483
55	-0.5947	-0.0664
60	-0.5374	-0.0815
64	-0.4710	-0.0875
70	-0.3925	-0.0845
75	-0.3110	-0.0785
80	-0.2294	-0.0634
85	-0.1449	-0.0423
90	-0.0634	-0.0121
95	0.0000	0.0091

	UPPER	LOWER	TOTAL
CN	0.7511	0.0776	0.8287
CC	-0.1028	0.0231	-0.0798
CM	-0.2157	0.0063	-0.2095

AIRFOIL PERFORMANCE

CL	CD	CM
0.8317	0.0363	-0.2095

TABLE 2.16

NACA SECTION ANALYSIS
0012-64

RUN NO. = 42

ALPHA = 6.0

MACH NO. =0.2987

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 5

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.6414	-0.6414
1	-2.2240	0.9411
2	-1.9632	0.7044
5	-1.6155	0.4676
7	-1.3158	0.3117
9	-1.1210	0.2368
15	-0.9681	0.1289
20	-0.8482	0.0659
25	-0.7793	0.0120
29	-0.7253	-0.0210
35	-0.6804	-0.0629
40	-0.6414	-0.0869
44	-0.6024	-0.1109
50	-0.5635	-0.1229
55	-0.5095	-0.1349
60	-0.4646	-0.1439
64	-0.4106	-0.1409
70	-0.3477	-0.1349
75	-0.2817	-0.1199
80	-0.2068	-0.0959
85	-0.1259	-0.0719
90	-0.0360	-0.0450
95	0.0480	-0.0270

	UPPER	LOWER	TOTAL
CN	0.5909	-0.0055	0.5854
CC	-0.0706	0.0224	-0.0482
CM	-0.1771	0.0357	-0.1413

AIRFOIL PERFORMANCE

CL	CD	CM
0.5872	0.0133	-0.1413

TABLE 2.17

NACA SECTION ANALYSIS
0012-64

RUN NO. = 42

ALPHA = 4.0

MACH NO. =0.3042

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 6

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.5558	-0.5558
1	-1.2824	0.7266
2	-1.2679	0.4342
5	-1.0624	0.2229
7	-0.9553	0.0984
9	-0.8656	0.0405
15	-0.7411	-0.0347
20	-0.6542	-0.0782
25	-0.6050	-0.1158
29	-0.5732	-0.1390
35	-0.5471	-0.1650
40	-0.5240	-0.1853
44	-0.5008	-0.1997
50	-0.4748	-0.1997
55	-0.4313	-0.2026
60	-0.4024	-0.2026
64	-0.3590	-0.1882
70	-0.3069	-0.1766
75	-0.2518	-0.1534
80	-0.1882	-0.1303
85	-0.1129	-0.1042
90	-0.0289	-0.0782
95	0.0695	0.0145

	UPPER	LOWER	TOTAL
CN	0.4576	-0.0931	0.3645
CC	-0.0457	0.0144	-0.0313
CM	-0.1456	0.0614	-0.0841

AIRFOIL PERFORMANCE

CL	CD	CM
0.3658	-0.0058	-0.0841

TABLE 2.18

NACA SECTION ANALYSIS
0012-64

RUN NO. = 41

ALPHA = 2.0

MACH NO. =0.2961

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 7

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.1715	-0.1715
1	-0.5115	0.3400
2	-0.6615	0.0521
5	-0.6309	-0.0796
7	-0.6033	-0.1501
9	-0.5635	-0.1838
15	-0.5206	-0.2144
20	-0.4808	-0.2358
25	-0.4594	-0.2573
29	-0.4471	-0.2634
35	-0.4380	-0.2726
40	-0.4288	-0.2818
44	-0.4165	-0.2818
50	-0.4012	-0.2756
55	-0.3736	-0.2664
60	-0.3583	-0.2573
64	-0.3338	-0.2389
70	-0.2818	-0.2144
75	-0.2297	-0.1899
80	-0.1715	-0.1654
85	-0.1103	-0.1317
90	-0.0337	-0.0061
95	0.0643	0.0704

	UPPER	LOWER	TOTAL
CN	0.3430	-0.1826	0.1604
CC	-0.0207	0.0042	-0.0165
CM	-0.1219	0.0823	-0.0396

AIRFOIL PERFORMANCE

CL	CD	CM
0.1608	-0.0109	-0.0396

TABLE 2.19

NACA SECTION ANALYSIS
0012-64

RUN NO. = 40

ALPHA = 0.0

MACH NO. =0.2934

WING DATA FILE NAME = *WING1.DAT
INPUT FILE NO. - 8

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.0507	-0.0571
1	0.1203	-0.2344
2	-0.1419	-0.4349
5	-0.2437	-0.4472
7	-0.2838	-0.4411
9	-0.2868	-0.4411
15	-0.3023	-0.4195
20	-0.2961	-0.4102
25	-0.2961	-0.4040
29	-0.3053	-0.3917
35	-0.3115	-0.3886
40	-0.3115	-0.3855
44	-0.3115	-0.3763
50	-0.3053	-0.3578
55	-0.2899	-0.3424
60	-0.2776	-0.3239
64	-0.2560	-0.2992
70	-0.2313	-0.2714
75	-0.2066	-0.2437
80	-0.1820	-0.1666
85	-0.1234	-0.0894
90	-0.0031	-0.0185
95	0.0709	0.0617

	UPPER	LOWER	TOTAL
CN	0.2225	-0.2920	-0.0695
CC	-0.0017	-0.0114	-0.0131
CM	-0.0915	0.1097	0.0182

AIRFOIL PERFORMANCE

CL	CD	CM
-0.0695	-0.0131	0.0182

TABLE 2.20

NACA SECTION ANALYSIS
0012-64

RUN NO. = 45

ALPHA = -2.0

MACH NO. =0.2972

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 3

	UPPER SURFACE	LOWER SURFACE
%CHORD	CP LOCAL	CP LOCAL
0	-0.1946	-0.1946
1	0.5343	-0.9236
2	0.2475	-0.9900
5	0.1177	-0.8391
7	-0.0121	-0.7456
9	-0.0604	-0.7123
15	-0.1207	-0.6309
20	-0.1479	-0.5856
25	-0.1660	-0.5584
29	-0.1871	-0.5222
35	-0.2022	-0.5041
40	-0.2143	-0.4950
44	-0.2243	-0.4648
50	-0.2294	-0.4316
55	-0.2203	-0.4015
60	-0.2173	-0.3743
64	-0.2053	-0.3350
70	-0.1871	-0.2898
75	-0.1630	-0.2415
80	-0.1328	-0.1781
85	-0.0875	-0.1147
90	-0.0241	-0.0241
95	0.0543	0.0664

	UPPER	LOWER	TOTAL
CN	0.1232	-0.4026	-0.2795
CC	0.0105	-0.0321	-0.0217
CM	-0.0649	0.1344	0.0695

AIRFOIL PERFORMANCE

CL	CD	CM
-0.2801	-0.0119	0.0695

TABLE 2.21

NACA SECTION ANALYSIS
0012-64

RUN NO. = 46

ALPHA = -4.0

MACH NO. =0.2959

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 4

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.4750	-0.4750
1	0.8575	-1.8075
2	0.5583	-1.6625
5	0.3979	-1.3325
7	0.2375	-1.0919
9	0.1604	-0.9870
15	0.0617	-0.8205
20	-0.0123	-0.7372
25	-0.0339	-0.7002
29	-0.0648	-0.6570
35	-0.0956	-0.6200
40	-0.1234	-0.5891
44	-0.1357	-0.5552
50	-0.1450	-0.5059
55	-0.1511	-0.4596
60	-0.1542	-0.4287
64	-0.1573	-0.3701
70	-0.1542	-0.3054
75	-0.1265	-0.2498
80	-0.0956	-0.1820
85	-0.0771	-0.1110
90	-0.0216	-0.0123
95	0.0463	0.0679

	UPPER	LOWER	TOTAL
CN	0.0315	-0.5170	-0.4856
CC	0.0191	-0.0583	-0.0392
CM	-0.0392	0.1569	0.1177

AIRFOIL PERFORMANCE

CL	CD	CM
-0.4871	-0.0052	0.1177

TABLE 2.22

NACA SECTION ANALYSIS
0012-64

RUN NO. = 47

ALPHA = -6.0

MACH NO. =0.3004

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 5

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-0.9486	-0.9486
1	1.0107	-2.9079
2	0.8097	-2.4912
5	0.7654	-2.0775
7	0.4374	-1.4303
9	0.3487	-1.3151
15	0.2305	-1.0905
20	0.1655	-0.9752
25	0.0827	-0.8777
29	0.0473	-0.8097
35	0.0059	-0.7683
40	-0.0118	-0.7063
44	-0.0325	-0.6561
50	-0.0650	-0.6058
55	-0.0739	-0.5467
60	-0.0827	-0.4965
64	-0.0857	-0.4167
70	-0.0887	-0.3576
75	-0.0768	-0.2837
80	-0.0680	-0.2098
85	-0.0443	-0.1300
90	-0.0089	-0.0414
95	0.0296	0.0207

	UPPER	LOWER	TOTAL
CN	-0.0623	-0.6746	-0.7369
CC	0.0232	-0.0911	-0.0679
CM	-0.0098	0.1942	0.1845

AIRFOIL PERFORMANCE

CL	CD	CM
-0.7399	0.0095	0.1845

TABLE 2.23

NACA SECTION ANALYSIS
0012-64

RUN NO. = 48

ALPHA = -8.0

MACH NO. =0.2959

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 6

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-1.4744	-1.4744
1	1.0215	-3.9703
2	0.9485	-3.5812
5	0.7585	-2.2010
7	0.5685	-1.8362
9	0.4682	-1.6842
15	0.3405	-1.3498
20	0.2523	-1.1765
25	0.1946	-1.0579
29	0.1398	-0.9576
35	0.0669	-0.8725
40	0.0456	-0.8056
44	0.0091	-0.7539
50	-0.0061	-0.6627
55	-0.0426	-0.6019
60	-0.0578	-0.5320
64	-0.0851	-0.4469
70	-0.0790	-0.3678
75	-0.0730	-0.2888
80	-0.0851	-0.2189
85	-0.0578	-0.1125
90	-0.0304	-0.0578
95	0.0000	-0.0122

	UPPER	LOWER	TOTAL
CN	-0.1001	-0.8053	-0.9054
CC	0.0225	-0.1234	-0.1009
CM	-0.0013	0.2199	0.2185

AIRFOIL PERFORMANCE

CL	CD	CM
-0.9106	0.0261	0.2185

TABLE 2.24

NACA SECTION ANALYSIS
0012-64

RUN NO. = 49

ALPHA = -10.0

MACH NO. =0.3013

WING DATA FILE NAME = *STWD.DAT
INPUT FILE NO. - 7

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE CP LOCAL
0	-1.9572	-1.9572
1	0.9913	-4.9058
2	1.0062	-4.8939
5	0.9228	-2.6881
7	0.6906	-2.1880
9	0.5775	-1.9677
15	0.4138	-1.5658
20	0.3275	-1.3455
25	0.2530	-1.2205
29	0.1905	-1.0746
35	0.1340	-0.9824
40	0.0625	-0.8811
44	0.0387	-0.7918
50	0.0000	-0.7055
55	-0.0268	-0.6102
60	-0.0625	-0.5269
64	-0.0893	-0.4376
70	-0.0923	-0.3453
75	-0.1027	-0.2620
80	-0.1131	-0.2114
85	-0.1161	-0.1786
90	-0.1191	-0.1429
95	-0.1191	-0.1280

	UPPER	LOWER	TOTAL
CN	-0.1119	-0.9335	-1.0454
CC	0.0241	-0.1559	-0.1318
CM	-0.0098	0.2460	0.2362

AIRFOIL PERFORMANCE

CL	CD	CM
-1.0524	0.0517	0.2362

FIGURE 1

NACA 0012-64

SECTION PRESSURE

DISTRIBUTIONS AND FORCES

FIGURE 1.1

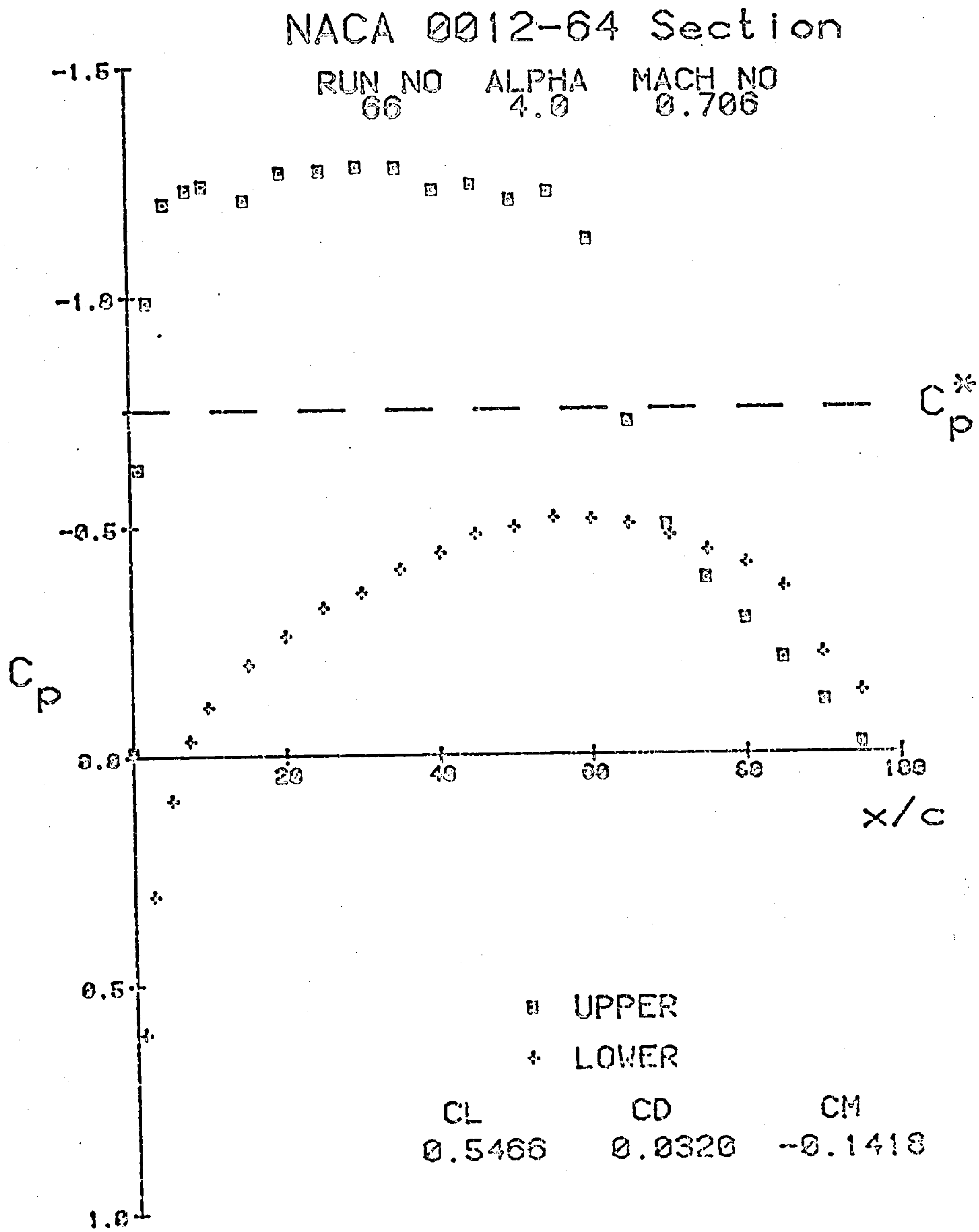

NACA 0012-64 Section
RUN NO 66
ALPHA 4.0
MACH NO 0.706
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.5466
CD 0.0320
CM -0.1418

FIGURE 1.2

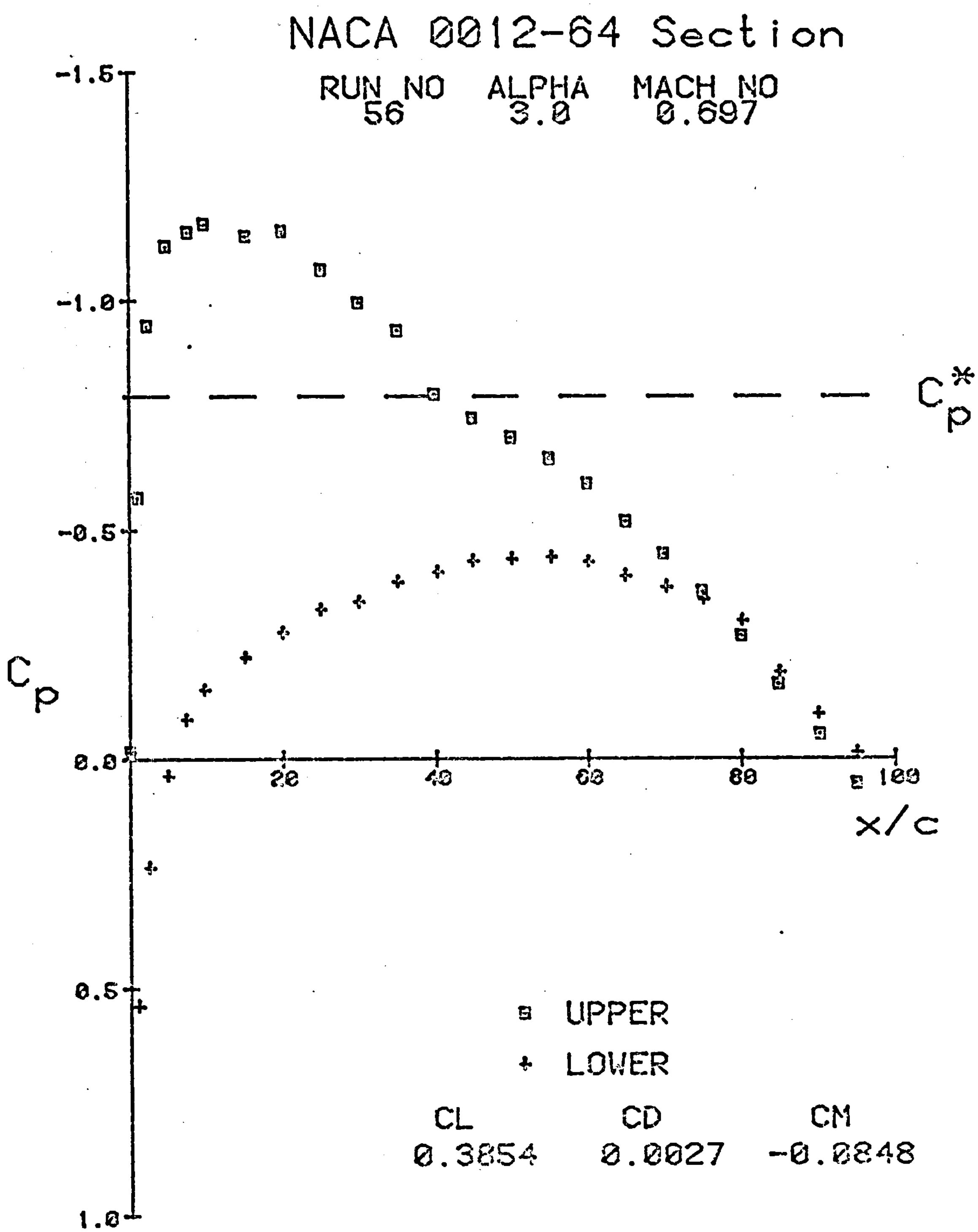

NACA 0012-64 Section
RUN NO 56
ALPHA 3.0
MACH NO 0.697
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.3854
CD 0.0027
CM -0.0848

FIGURE 1.3

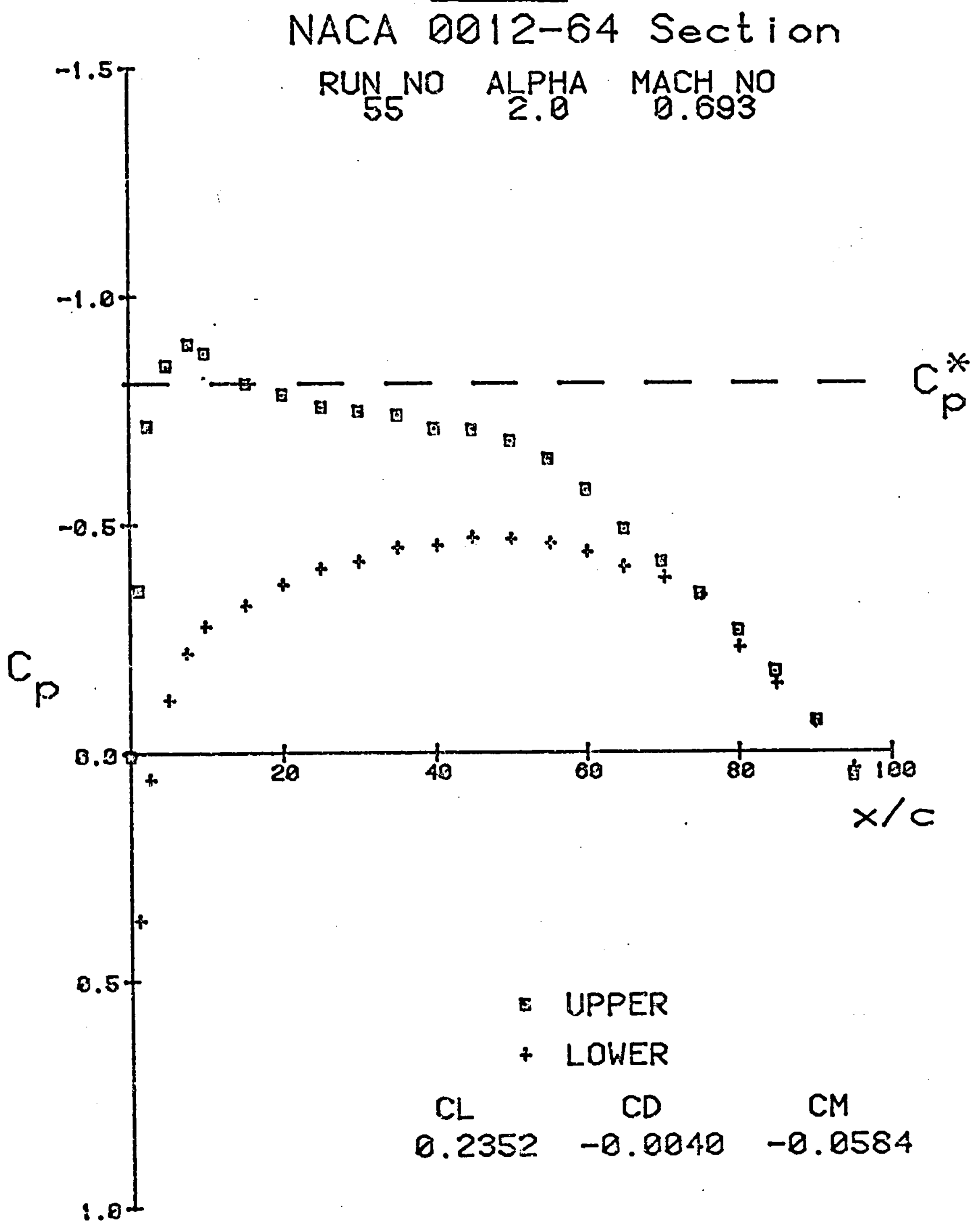

NACA 0012-64 Section
RUN NO 55
ALPHA 2.0
MACH NO 0.693
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
Cp*
UPPER
LOWER
CL 0.2352
CD -0.0040
CM -0.0584

FIGURE 1.4

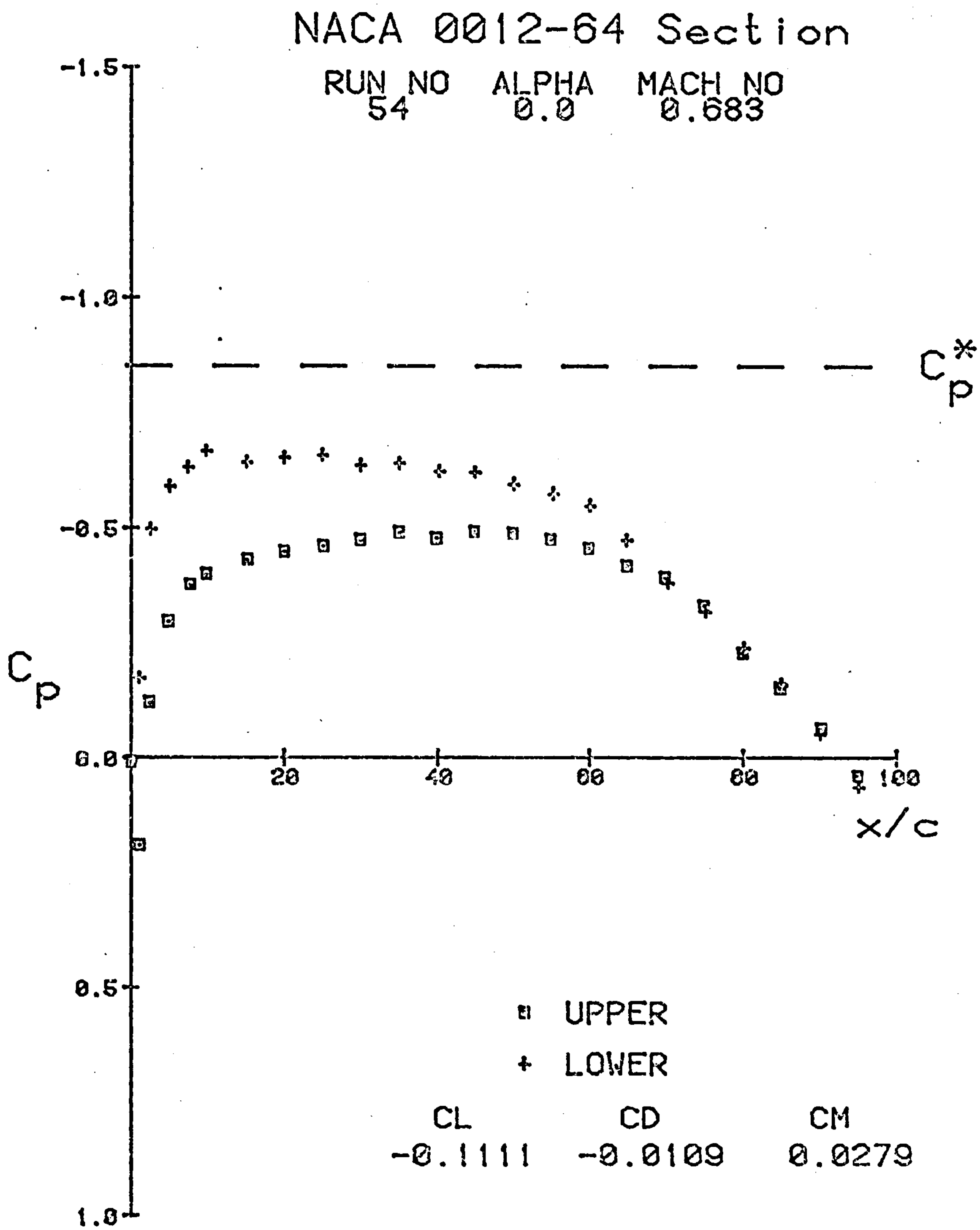

NACA 0012-64 Section
RUN NO 54
ALPHA 0.0
MACH NO 0.683
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.1111
CD -0.0109
CM 0.0279

FIGURE 1.5

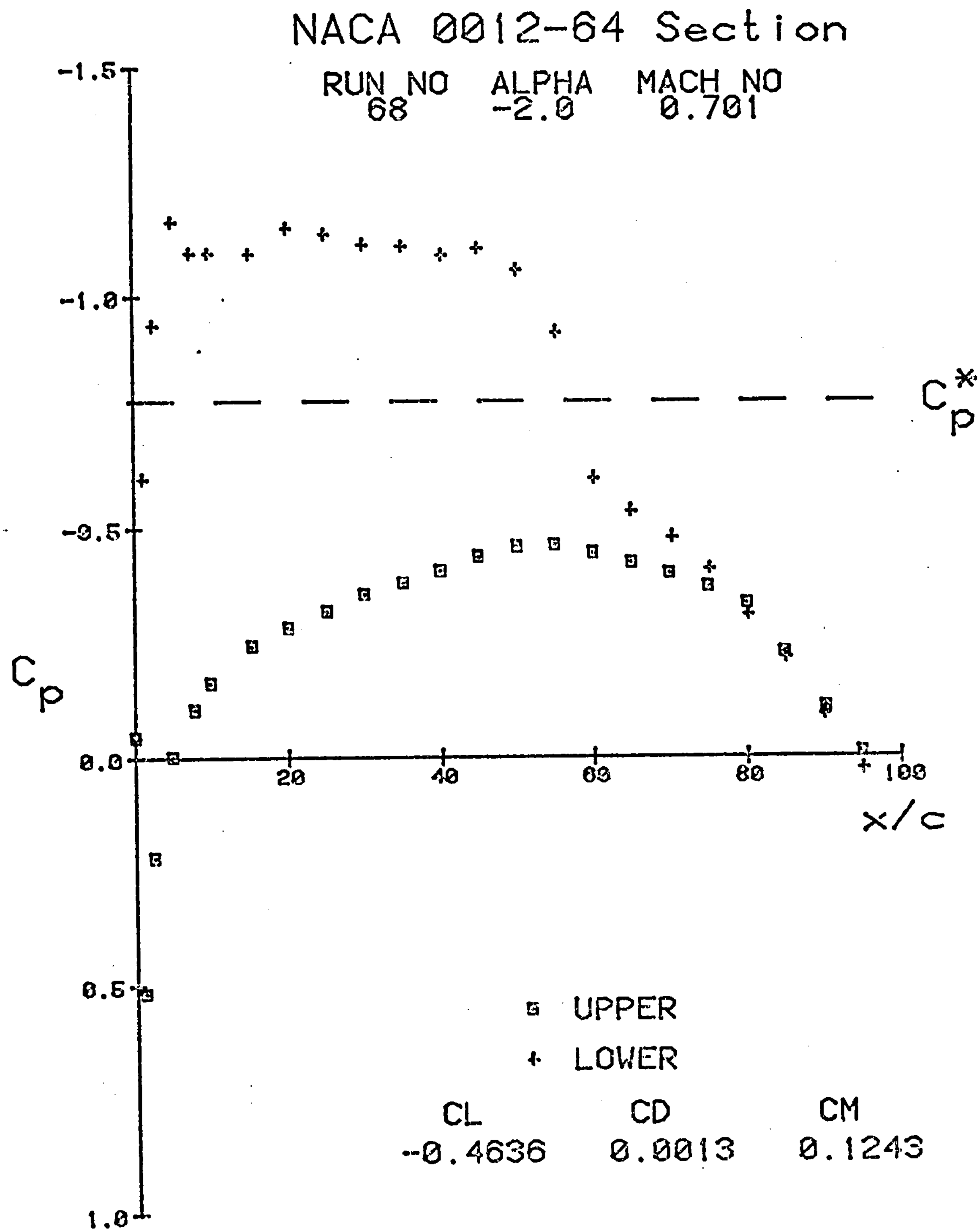
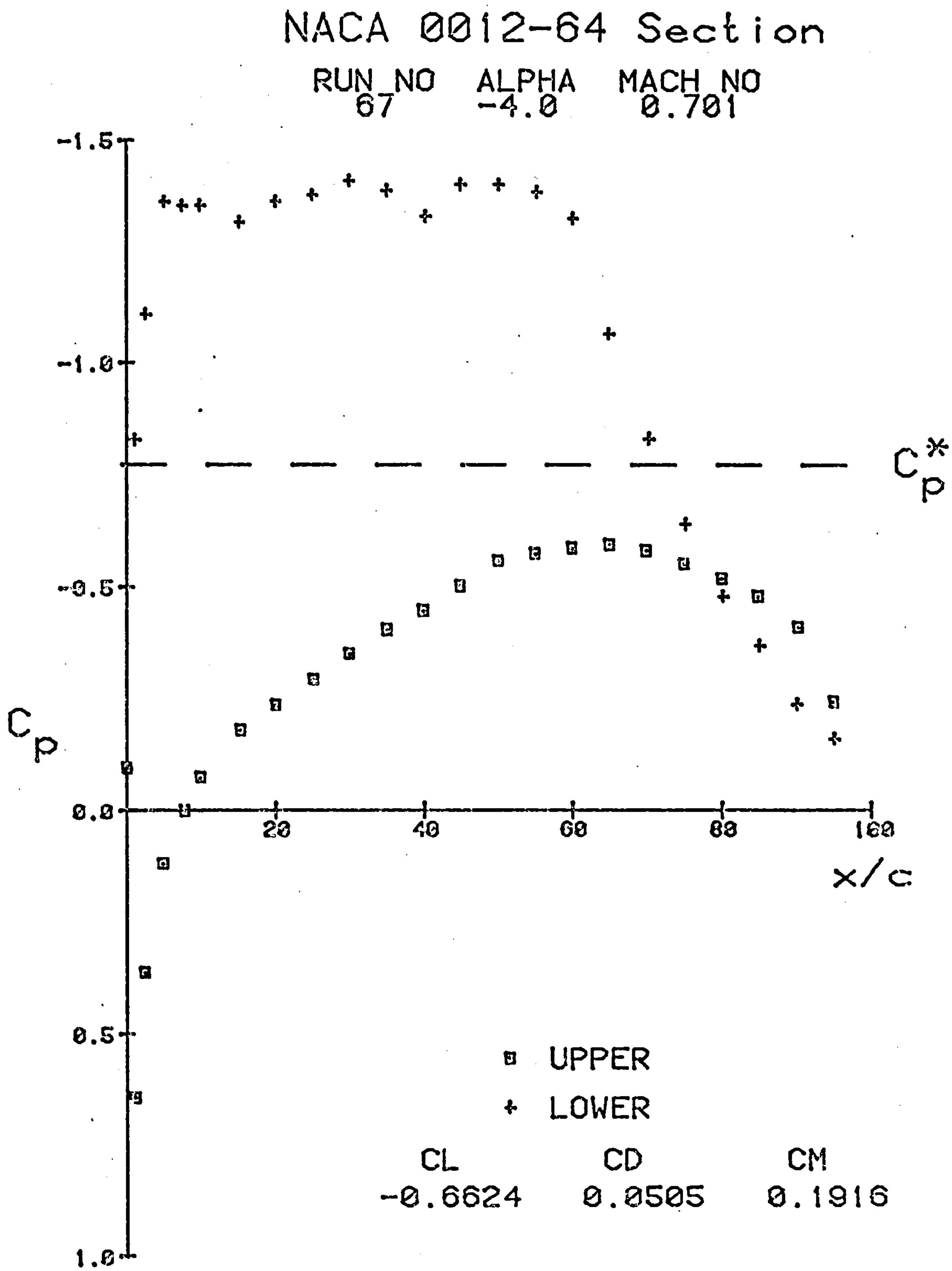

NACA 0012-64 Section
RUN NO 68
ALPHA -2.0
MACH NO 0.701
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p^*
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.4636
CD 0.0013
CM 0.1243

FIGURE 1.6



NACA 0012-64 Section
RUN NO 67
ALPHA -4.0
MACH NO 0.701
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
C_p
C_p^*
x/c
UPPER
LOWER
CL -0.6624
CD 0.0505
CM 0.1916

FIGURE 1.7

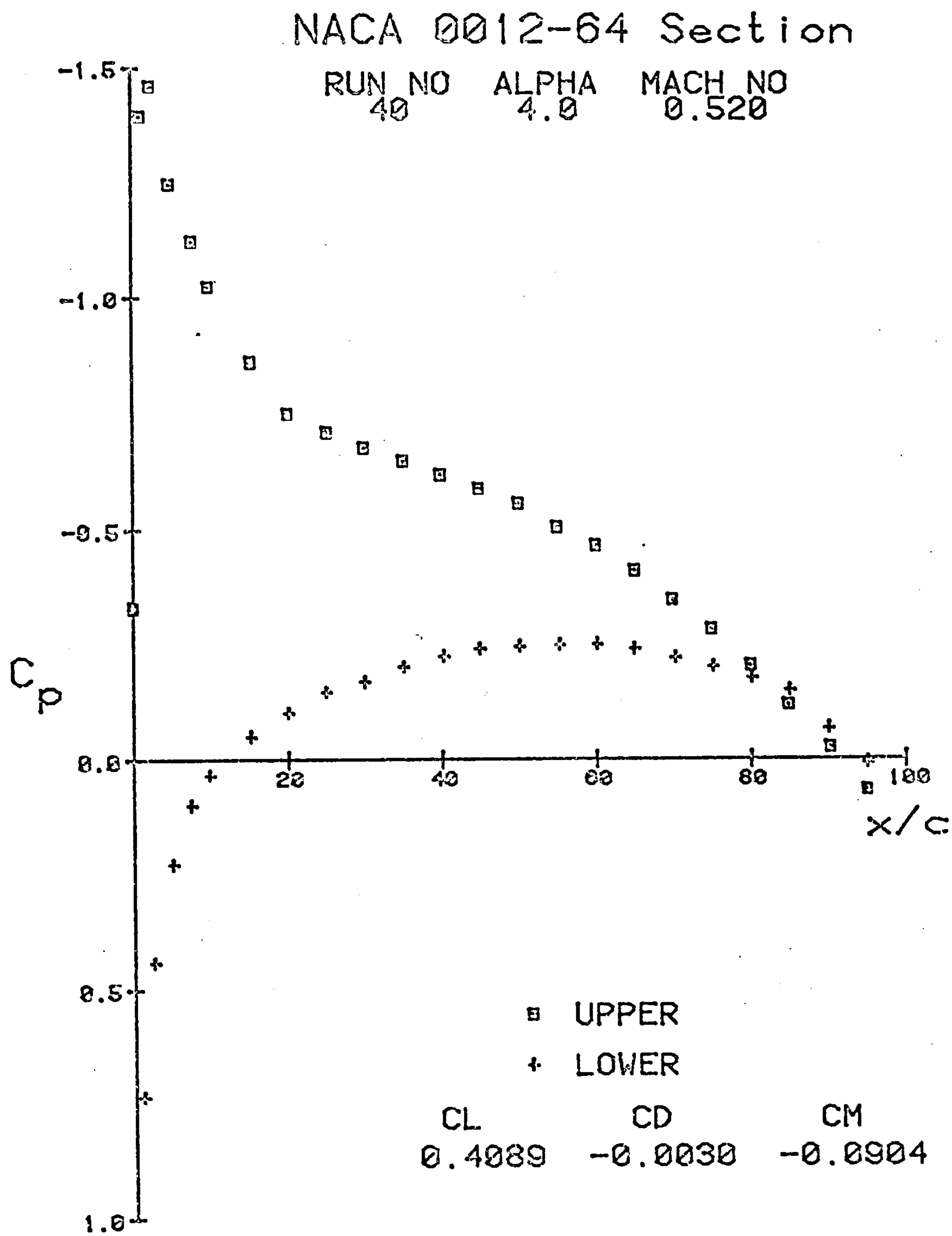
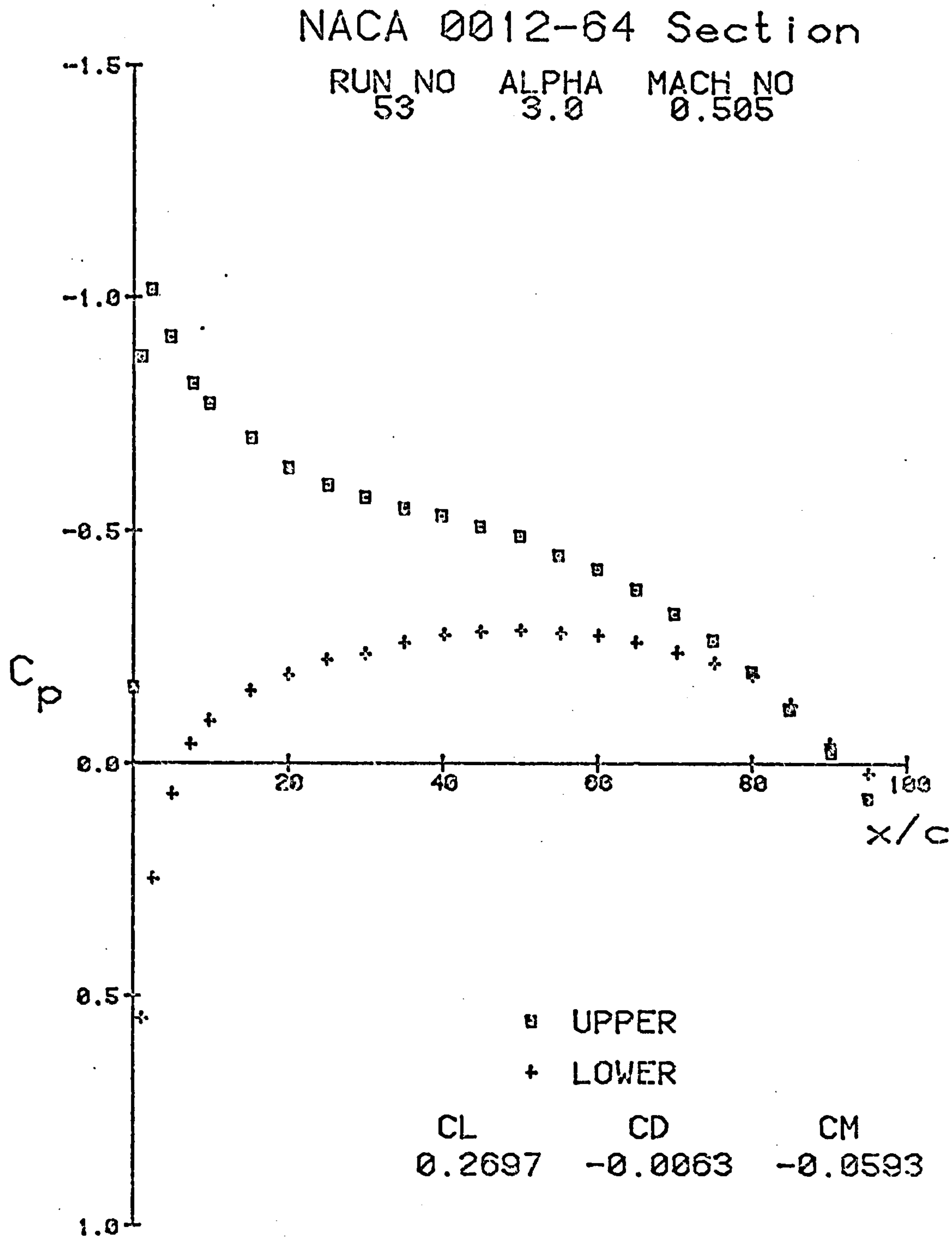

NACA 0012-64 Section
RUN NO 40
ALPHA 4.0
MACH NO 0.520
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4089
CD -0.0030
CM -0.0904

FIGURE 1.8



NACA 0012-64 Section
RUN NO 53
ALPHA 3.0
MACH NO 0.505
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.2697
CD -0.0063
CM -0.0593

FIGURE 1.9

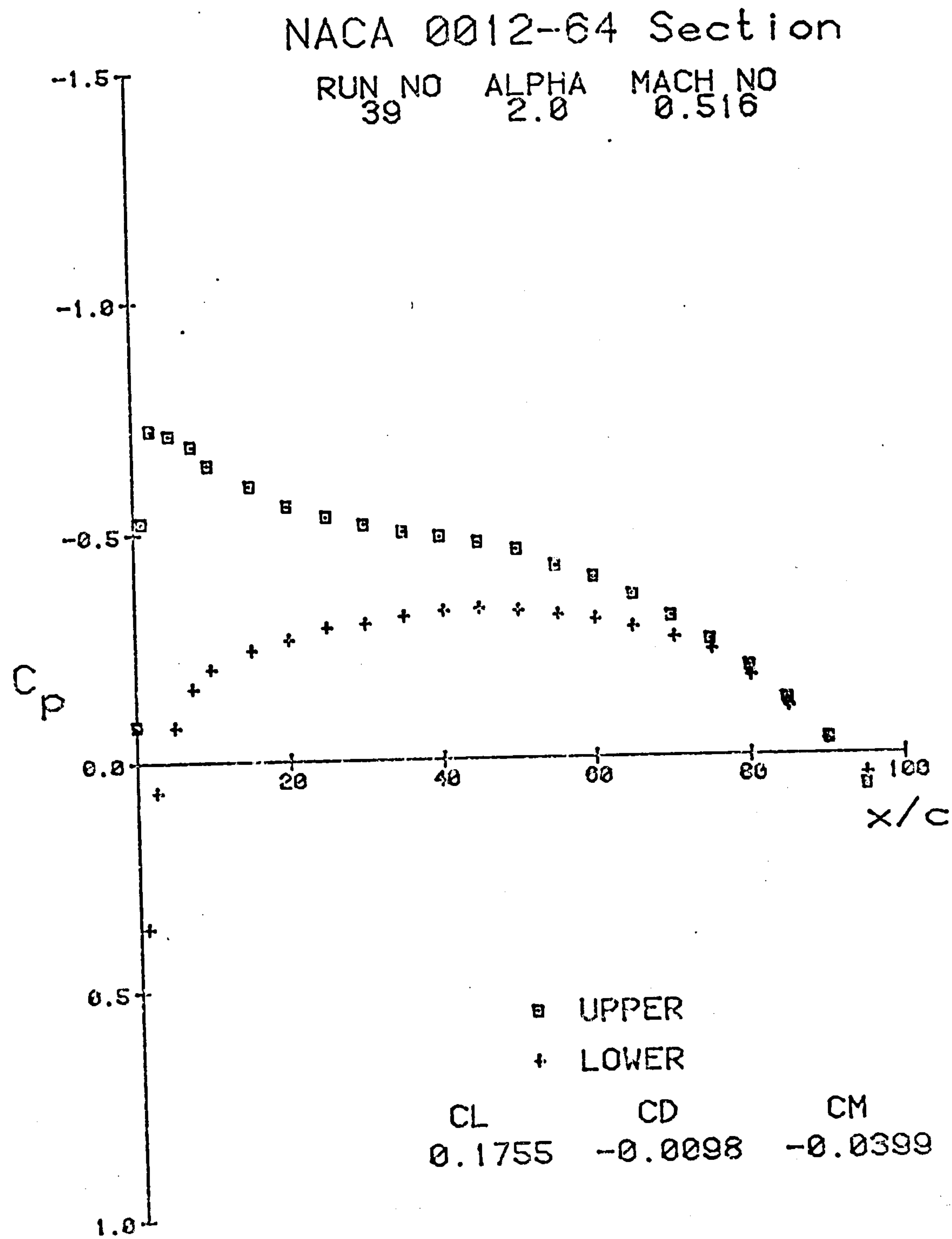
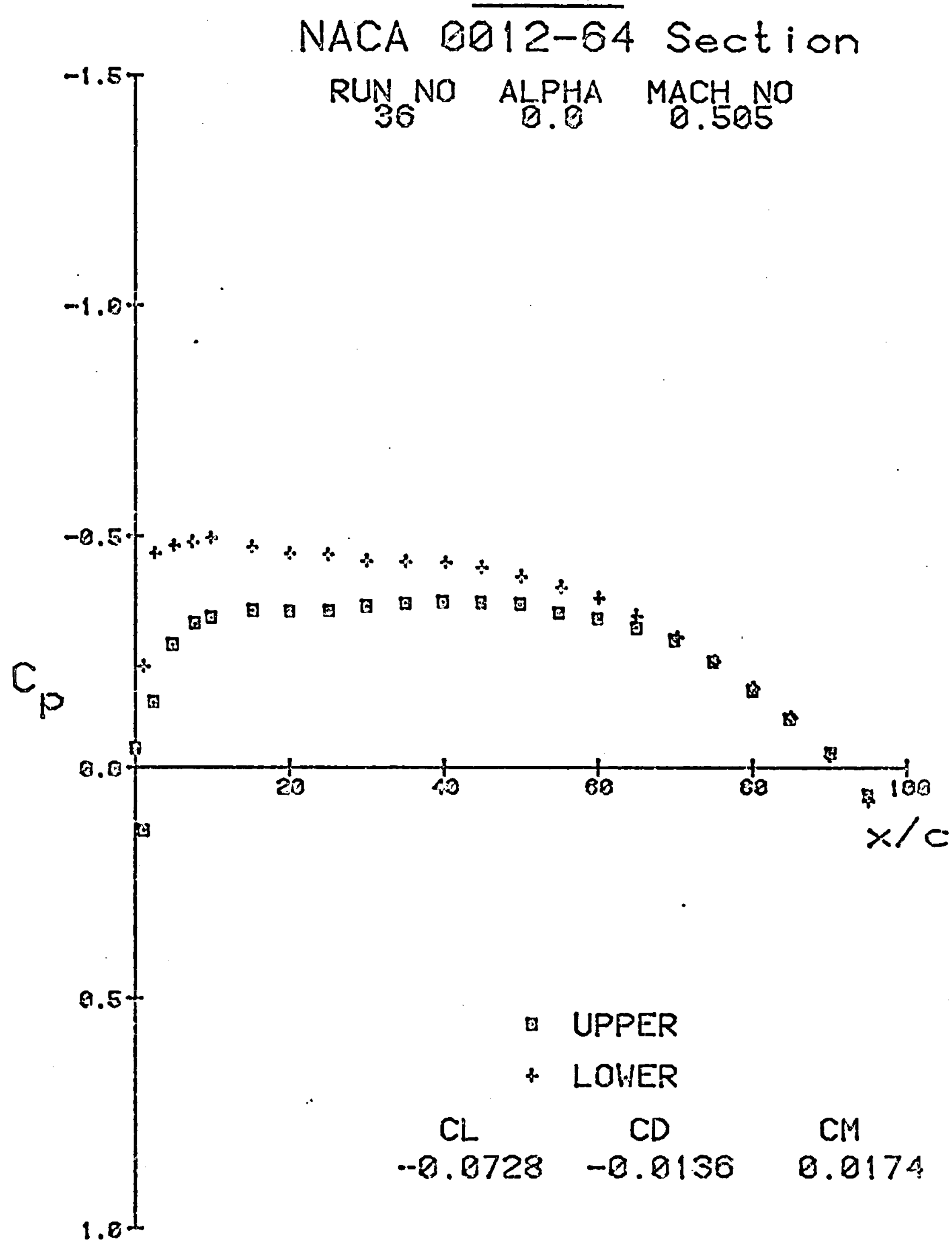

NACA 0012-64 Section
RUN NO 39
ALPHA 2.0
MACH NO 0.516
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1755
CD -0.0098
CM -0.0399

FIGURE 1.10



NACA 0012-64 Section
RUN NO 36
ALPHA 0.0
MACH NO 0.505
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.0728
CD -0.0136
CM 0.0174

FIGURE 1.11

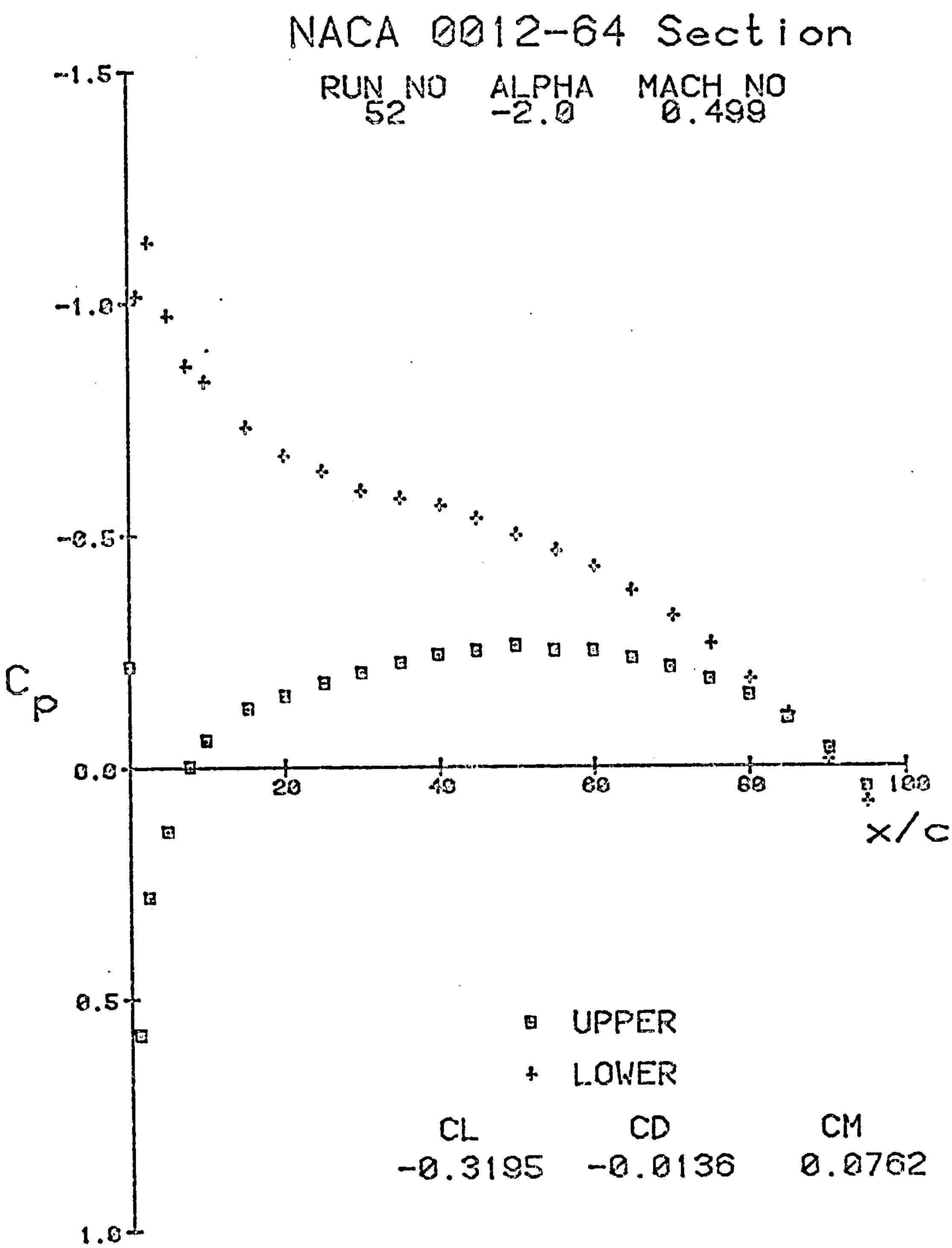

NACA 0012-64 Section
RUN NO 52
ALPHA -2.0
MACH NO 0.499
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.3195
CD -0.0136
CM 0.0762

FIGURE 1.12

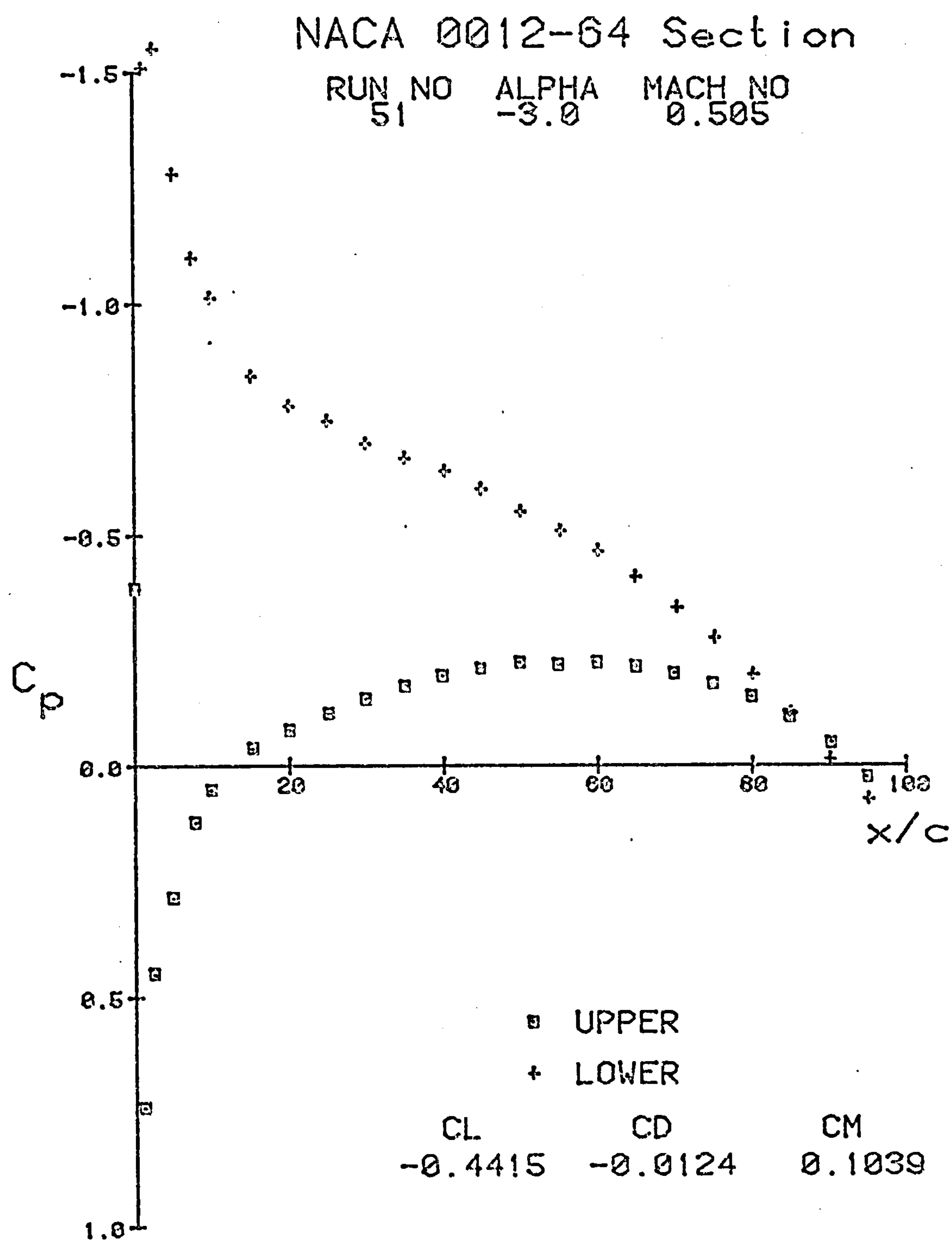

NACA 0012-64 Section
RUN NO 51
ALPHA -3.0
MACH NO 0.505
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.4415
CD -0.0124
CM 0.1039

FIGURE 1.13

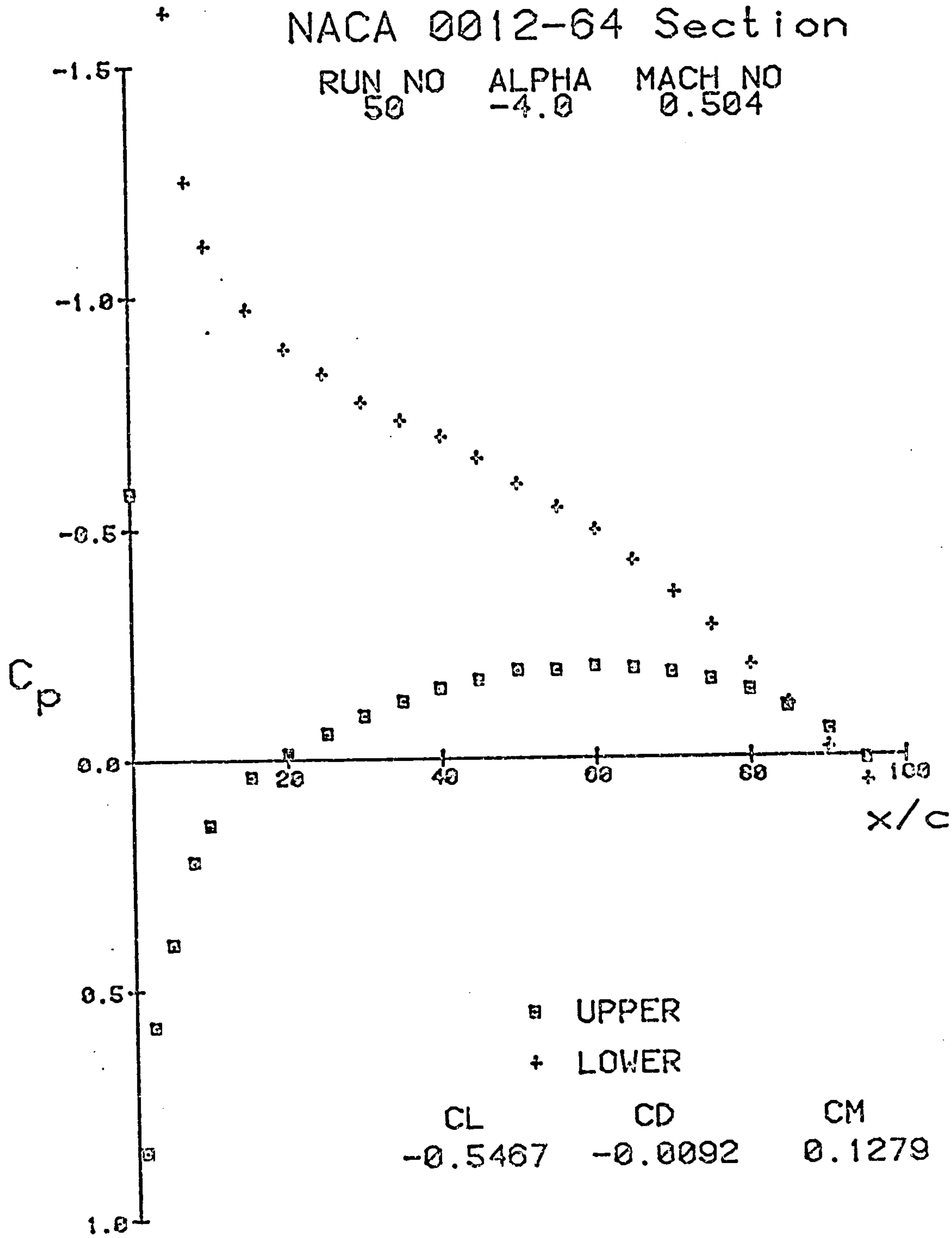

NACA 0012-64 Section
RUN NO 50
ALPHA -4.0
MACH NO 0.504
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.5467
CD -0.0092
CM 0.1279

FIGURE 1.14

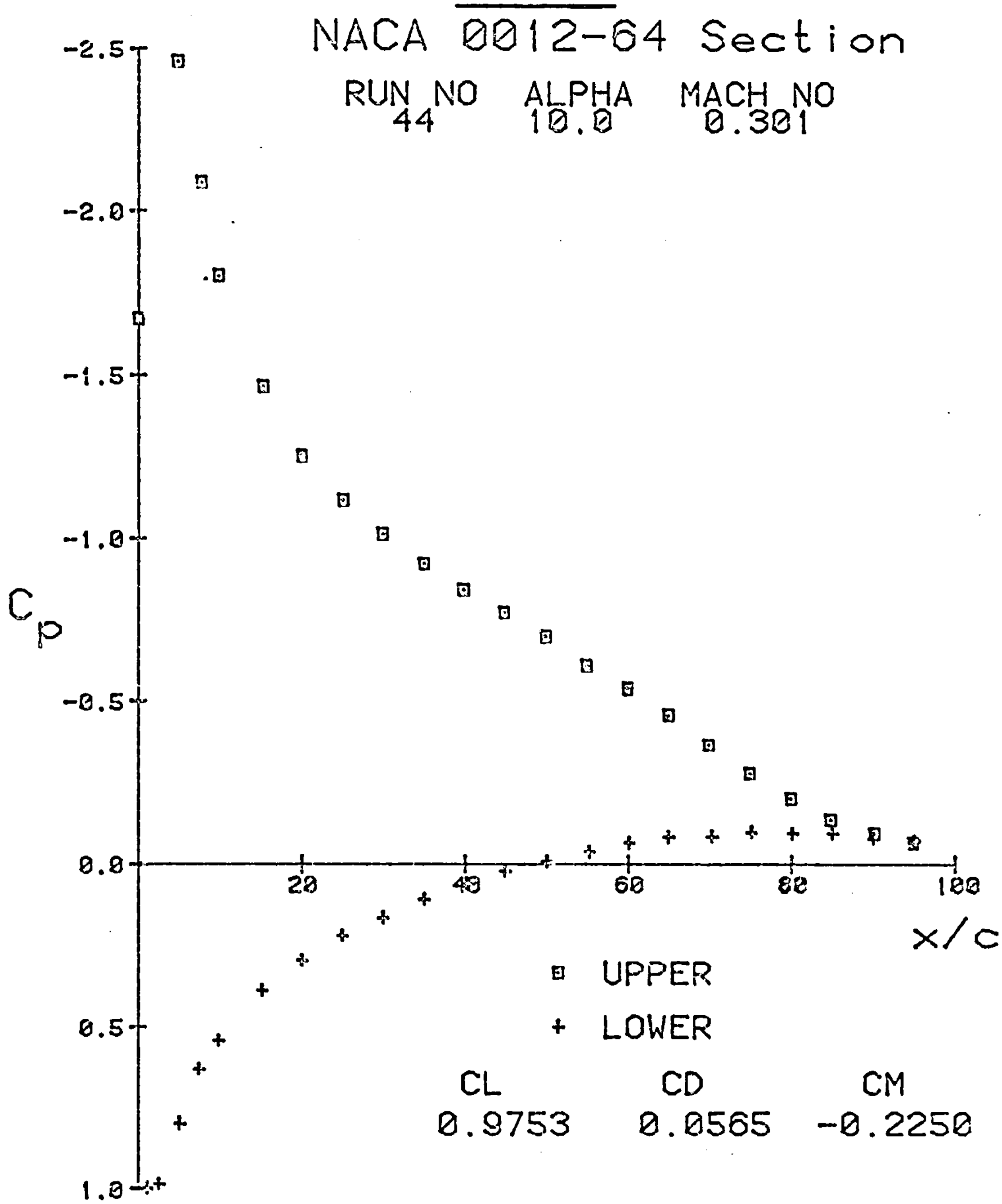

NACA 0012-64 Section
RUN NO 44
ALPHA 10.0
MACH NO 0.301
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.9753
CD 0.0565
CM -0.2250

FIGURE 1.15

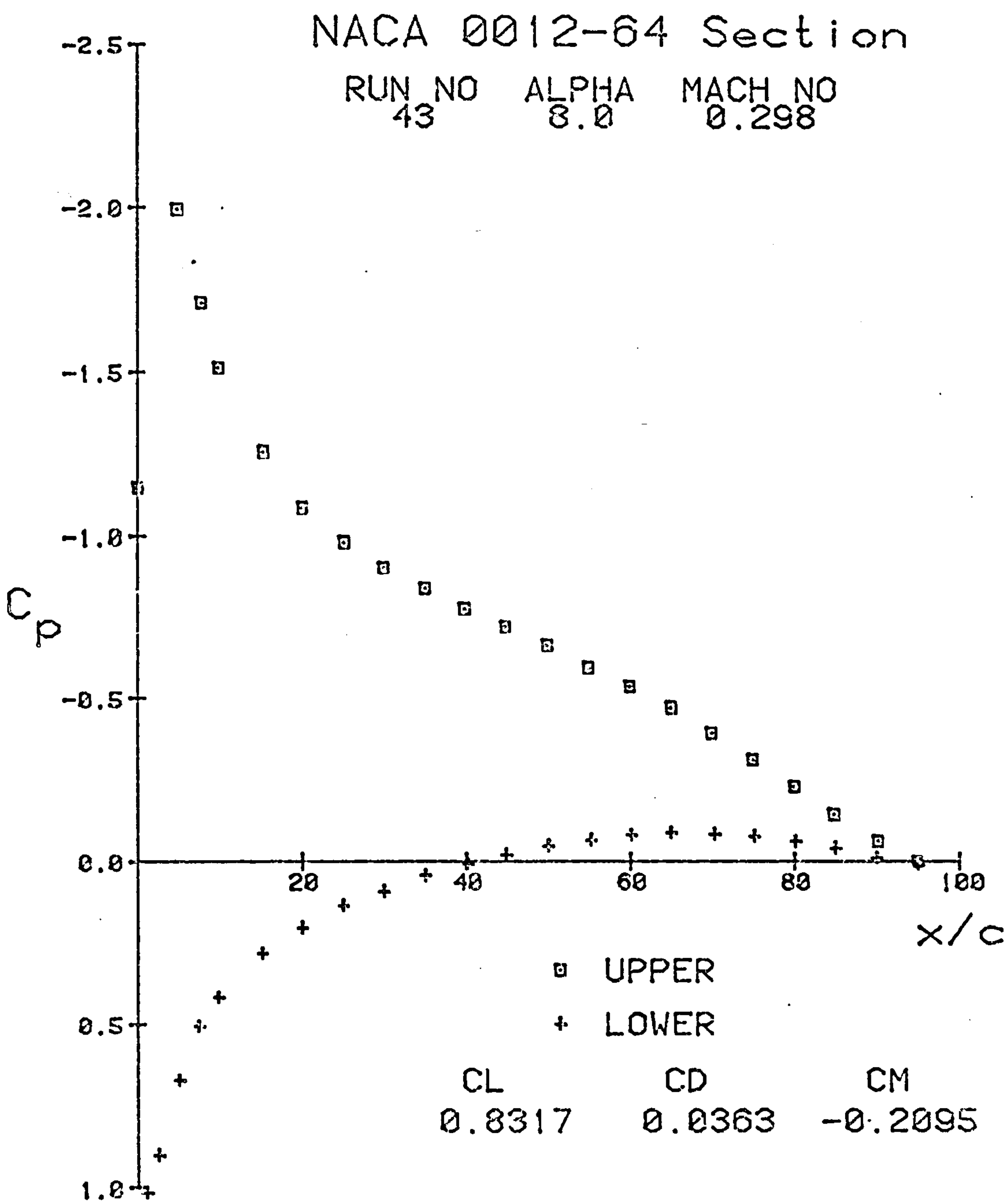
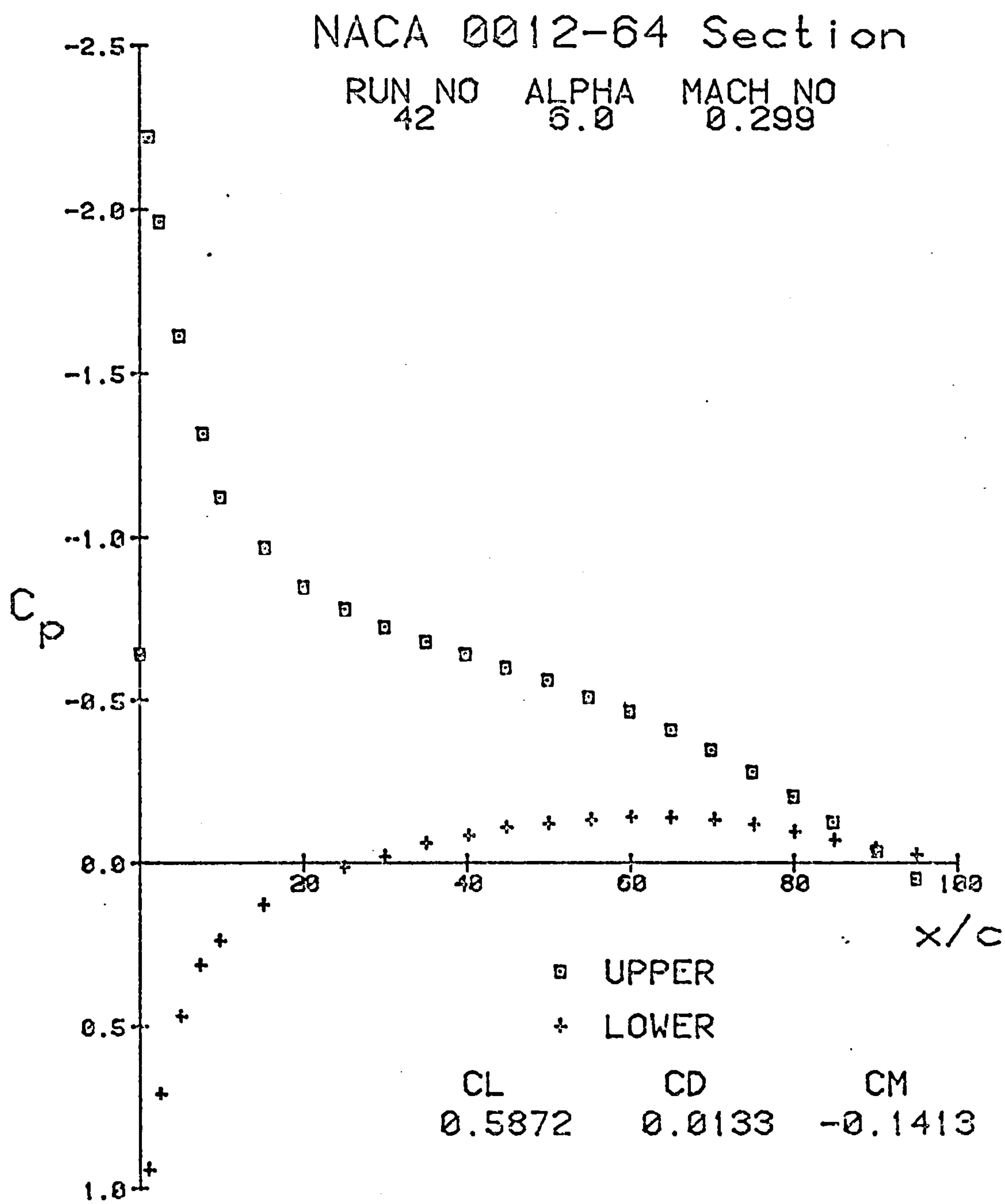

NACA 0012-64 Section
RUN NO 43
ALPHA 8.0
MACH NO 0.298
-2.5
-2.0
-1.5
-1.0
Cp
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.8317
CD 0.0363
CM -0.2095

FIGURE 1.16



NACA 0012-64 Section
RUN NO 42
ALPHA 6.0
MACH NO 0.299
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.5872
CD 0.0133
CM -0.1413

FIGURE 1.17

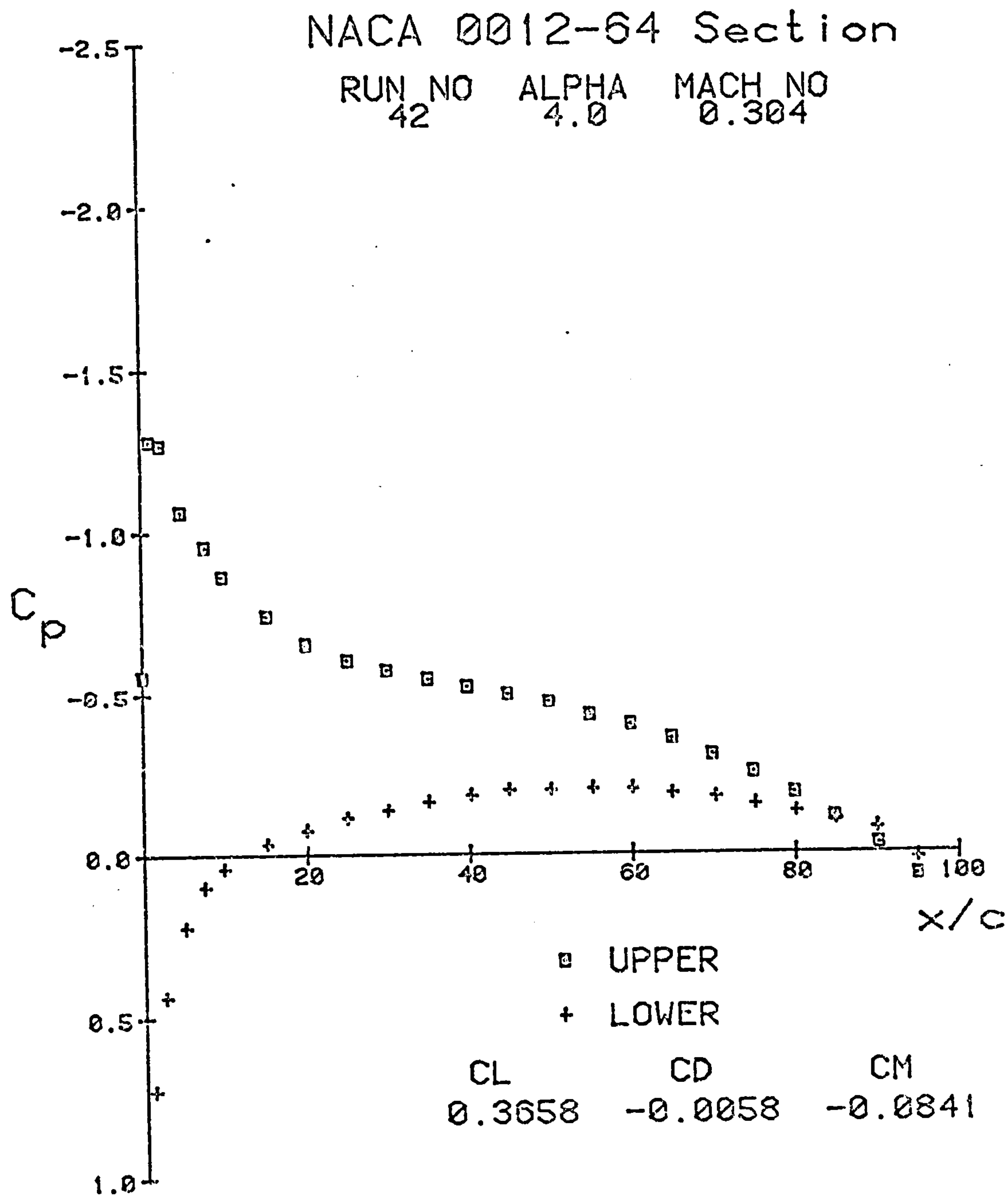

NACA 0012-64 Section
RUN NO 42
ALPHA 4.0
MACH NO 0.304
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.3658
CD -0.0058
CM -0.0841

FIGURE 1.18

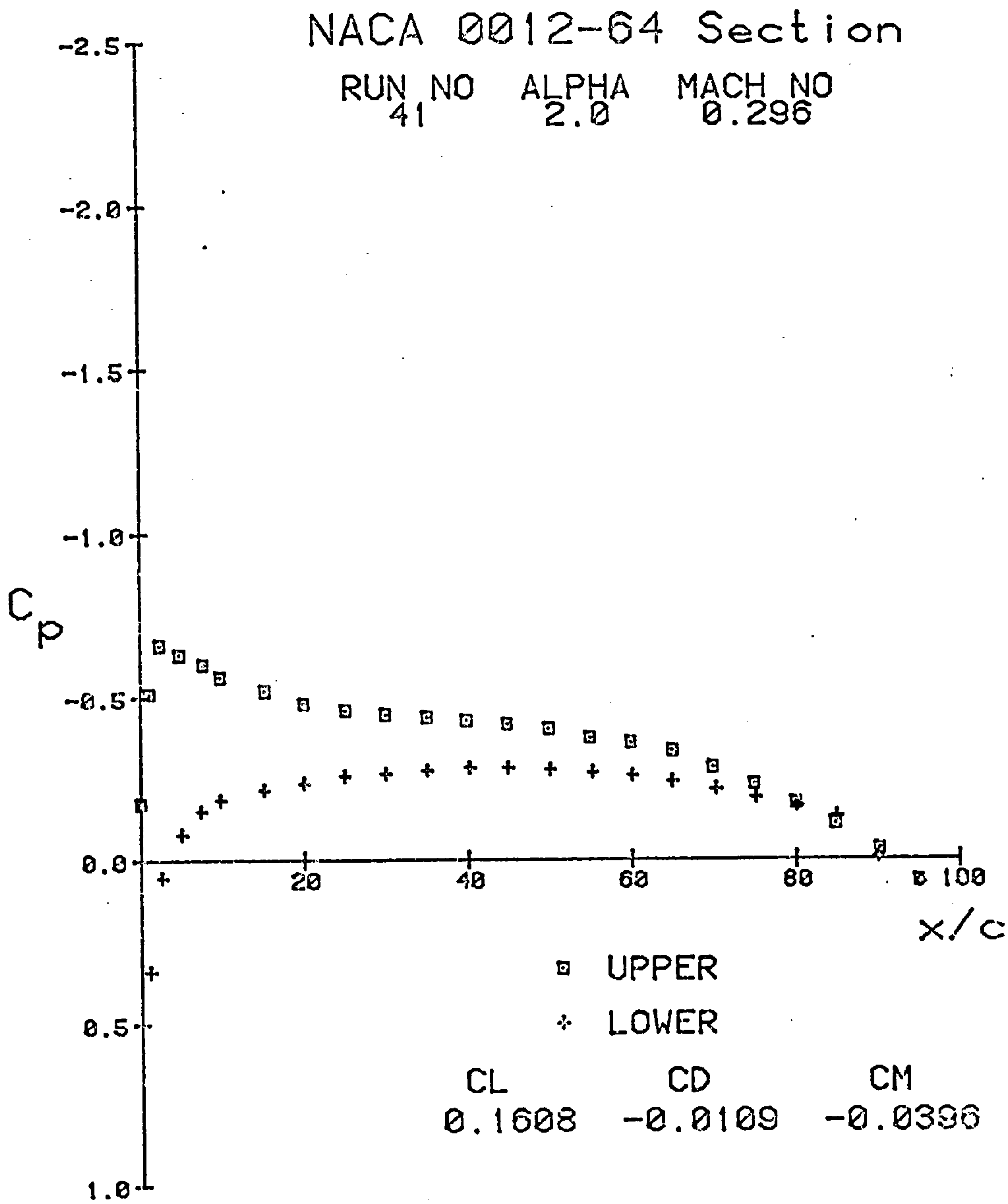
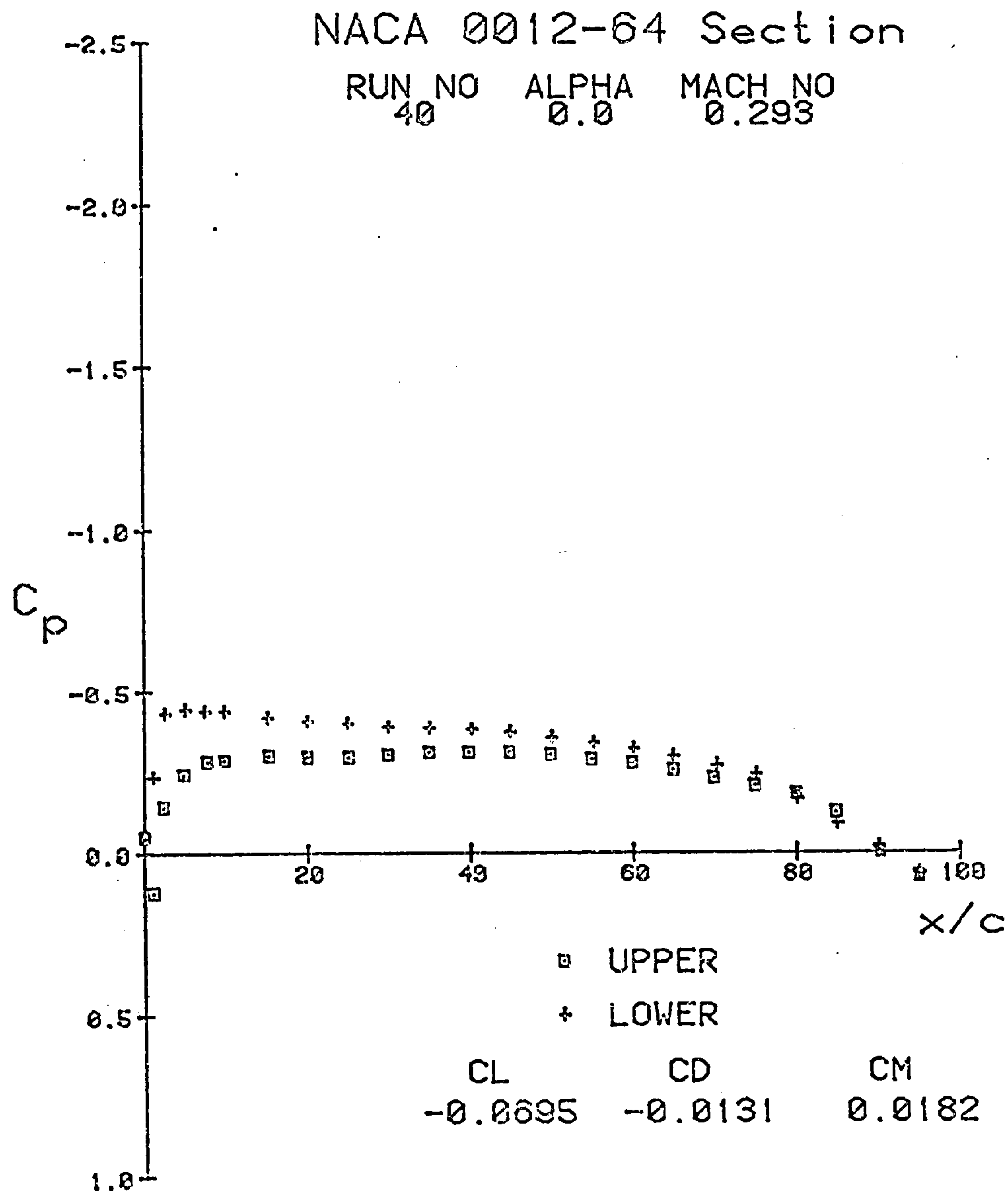

NACA 0012-64 Section
RUN NO 41
ALPHA 2.0
MACH NO 0.296
Cp
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1608
CD -0.0109
CM -0.0396

FIGURE 1.19



NACA 0012-64 Section
RUN NO 40
ALPHA 0.0
MACH NO 0.293
-2.5
-2.0
-1.5
-1.0
Cp
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.0695
CD -0.0131
CM 0.0182

FIGURE 1.20

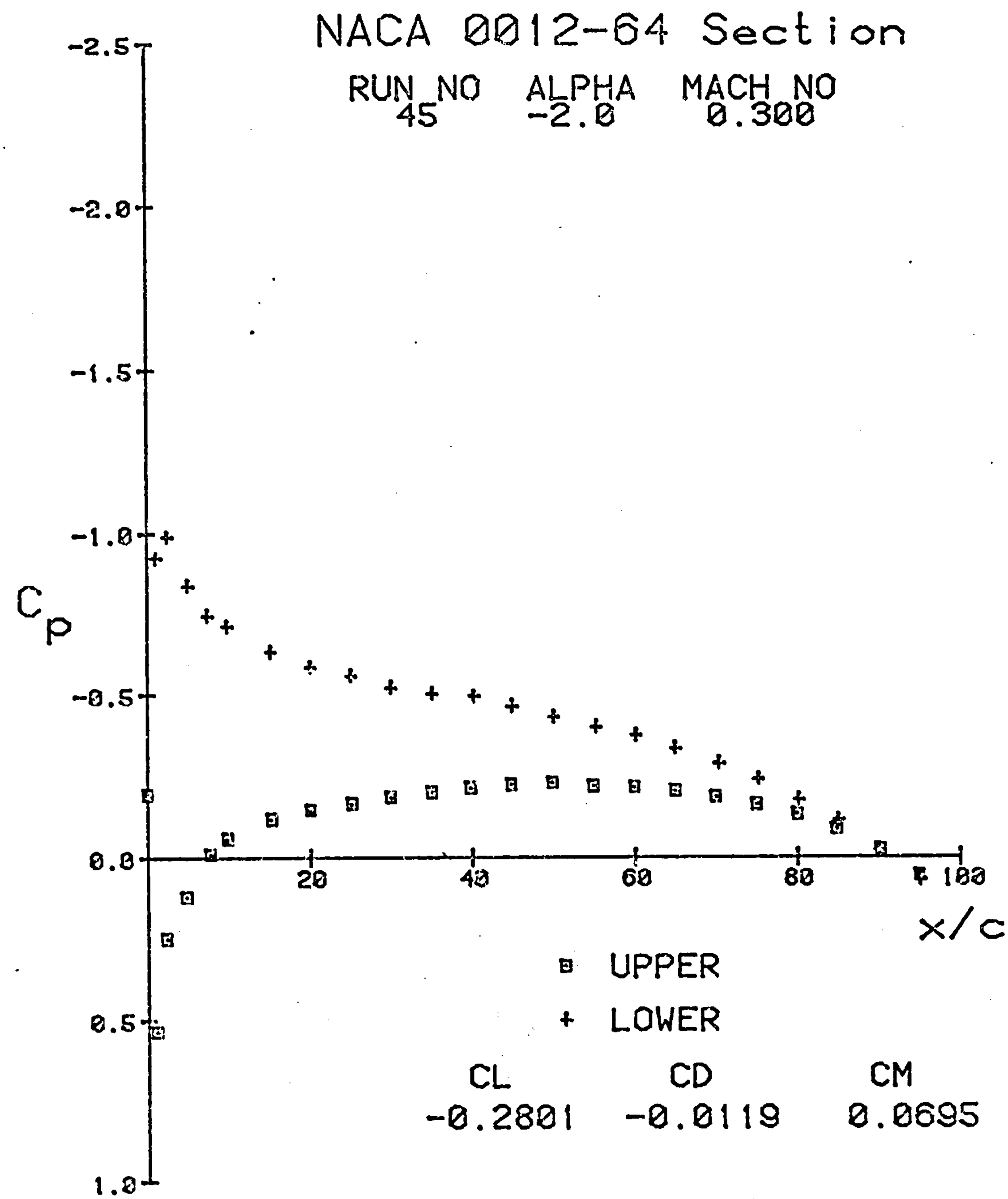

NACA 0012-64 Section
RUN NO 45
ALPHA -2.0
MACH NO 0.300
-2.5
-2.0
-1.5
-1.0
Cp
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.2801
CD -0.0119
CM 0.0695

FIGURE 1.21

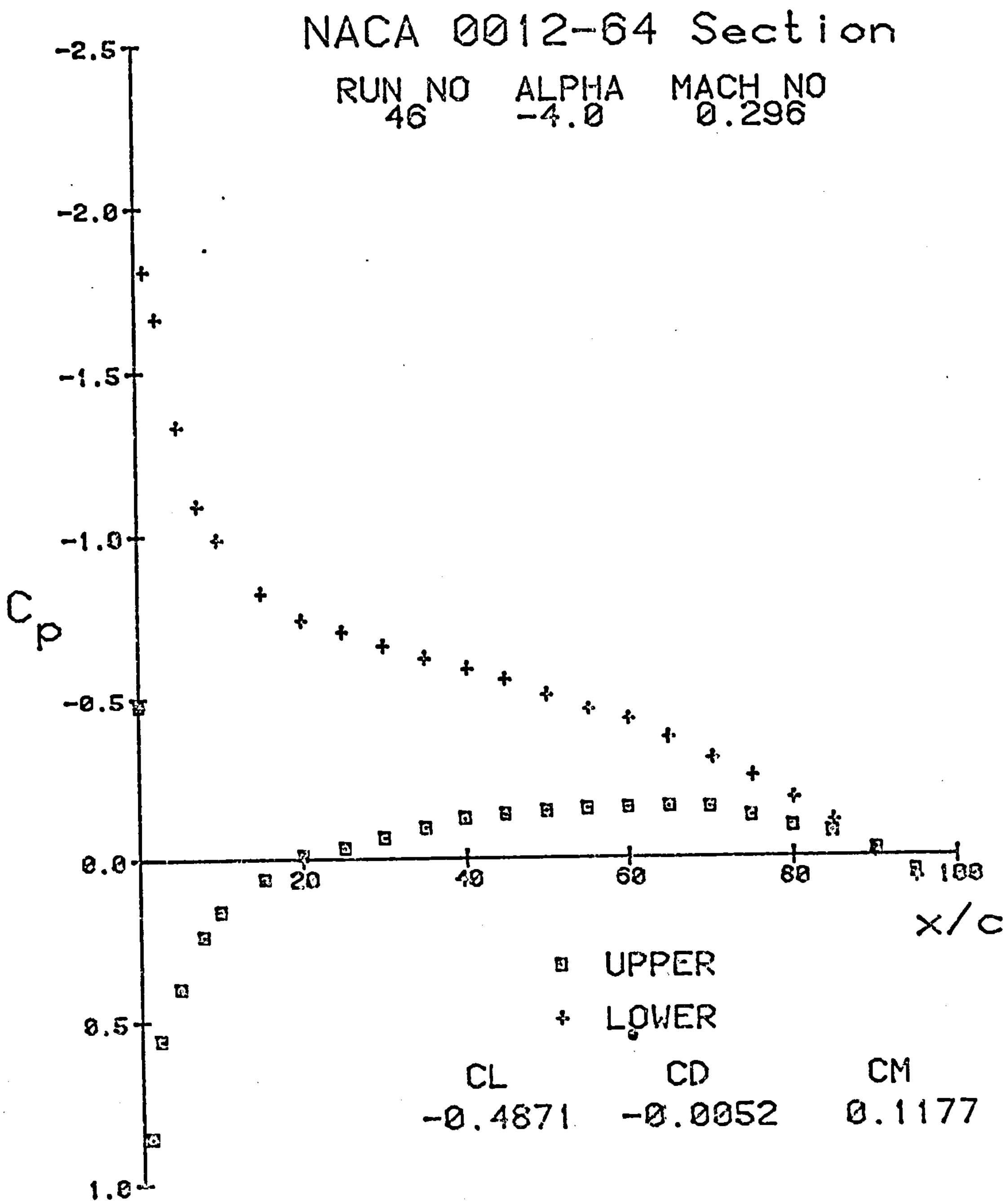

NACA 0012-64 Section
RUN NO 46
ALPHA -4.0
MACH NO 0.296
-2.5
-2.0
-1.5
-1.0
Cp
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.4871
CD -0.0052
CM 0.1177

FIGURE 1.22

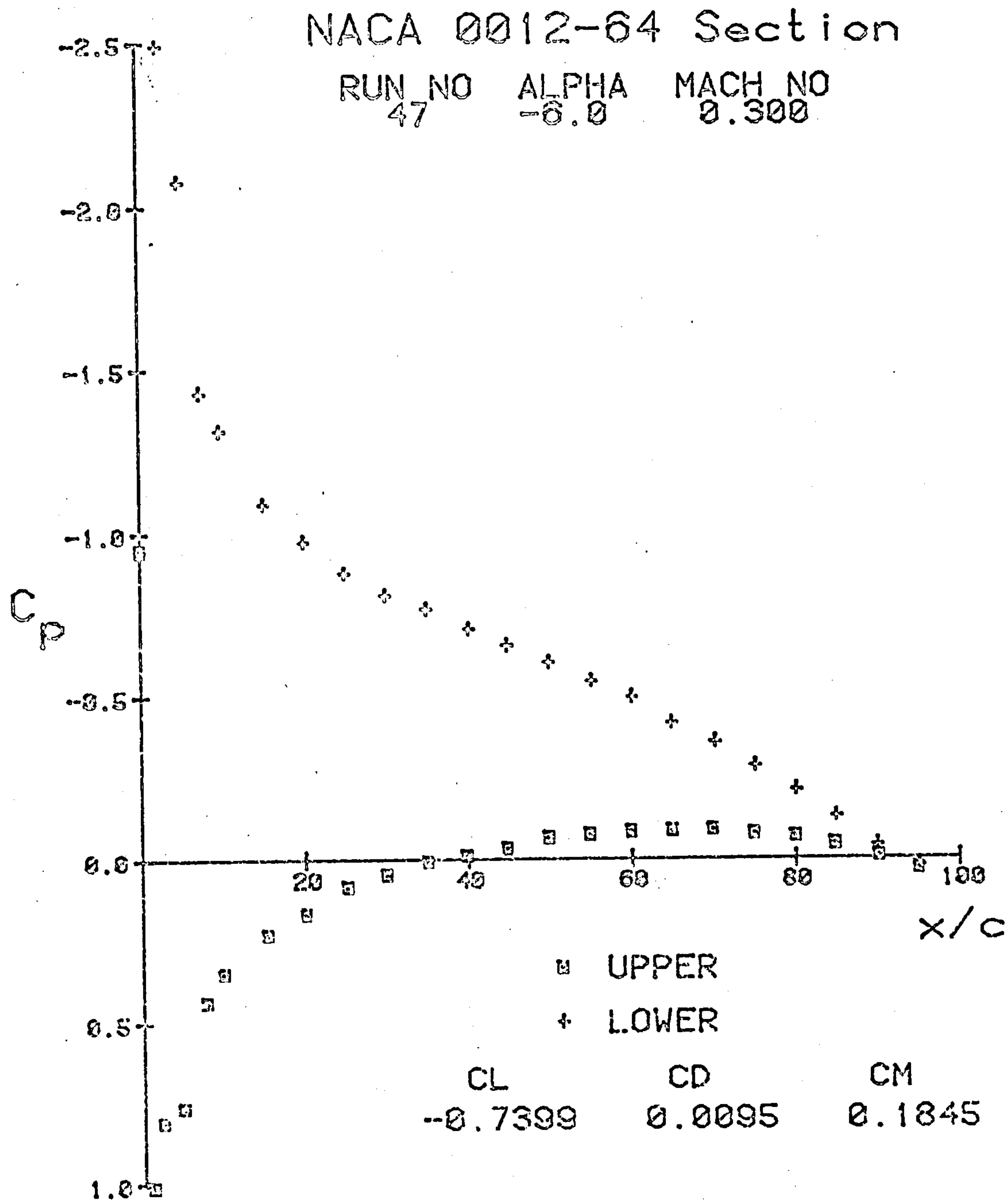

NACA 0012-64 Section
RUN NO 47
ALPHA -6.0
MACH NO 0.300
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.7399
CD 0.0095
CM 0.1845

FIGURE 1.23

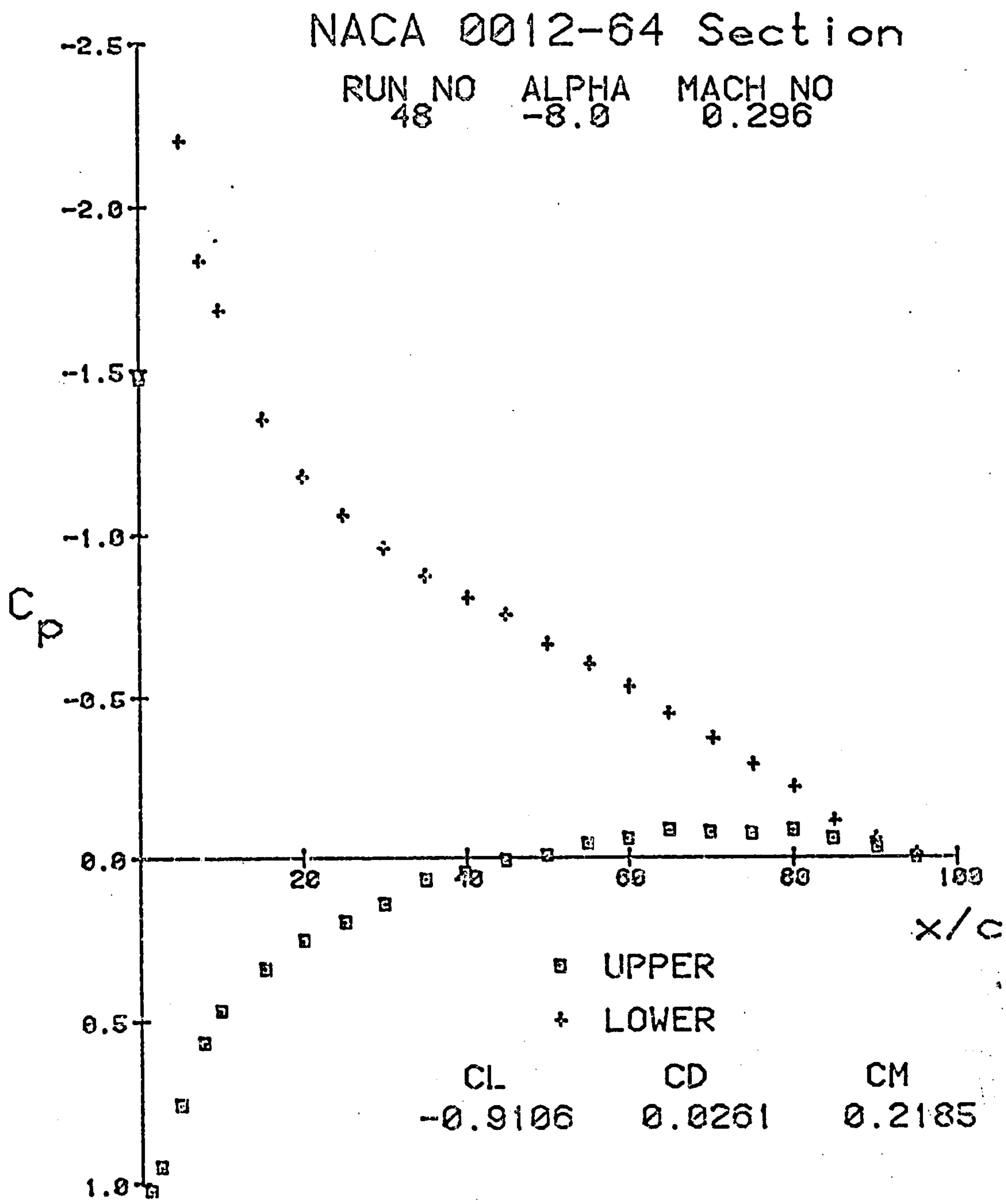

NACA 0012-64 Section
RUN NO 48
ALPHA -8.0
MACH NO 0.296
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL -0.9106
CD 0.0261
CM 0.2185

FIGURE 1.24

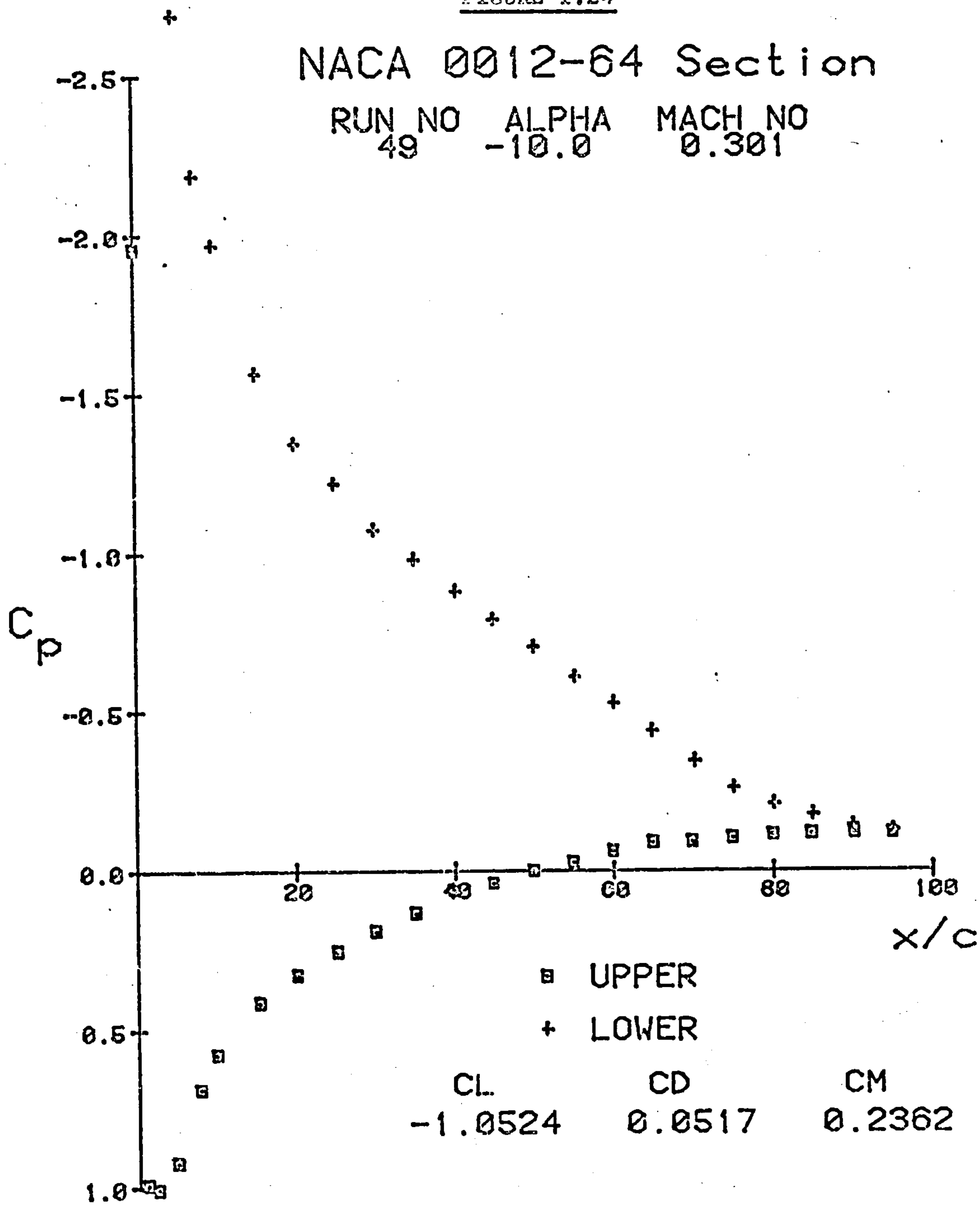

NACA 0012-64 Section
RUN NO ALPHA MACH NO
49 -10.0 0.301
-2.5
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL CD CM
-1.0524 0.0517 0.2362

FIGURE 2

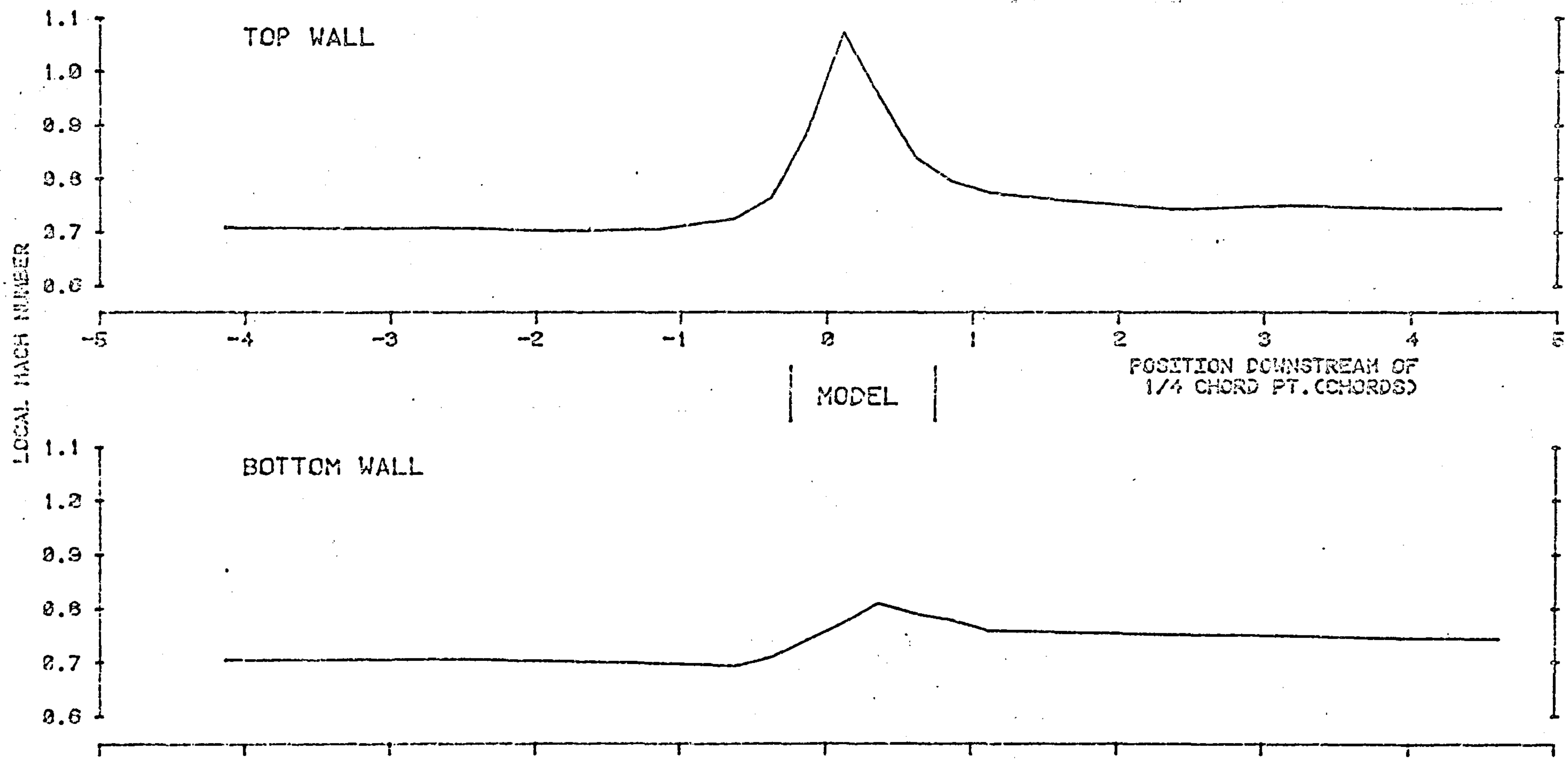
MACH NUMBER DISTRIBUTIONS ALONG

THE CENTRELINE OF EACH FLEXIBLE WALL

FIGURE 2.1

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 66 ALPHA 4.0 MACH NO 0.706

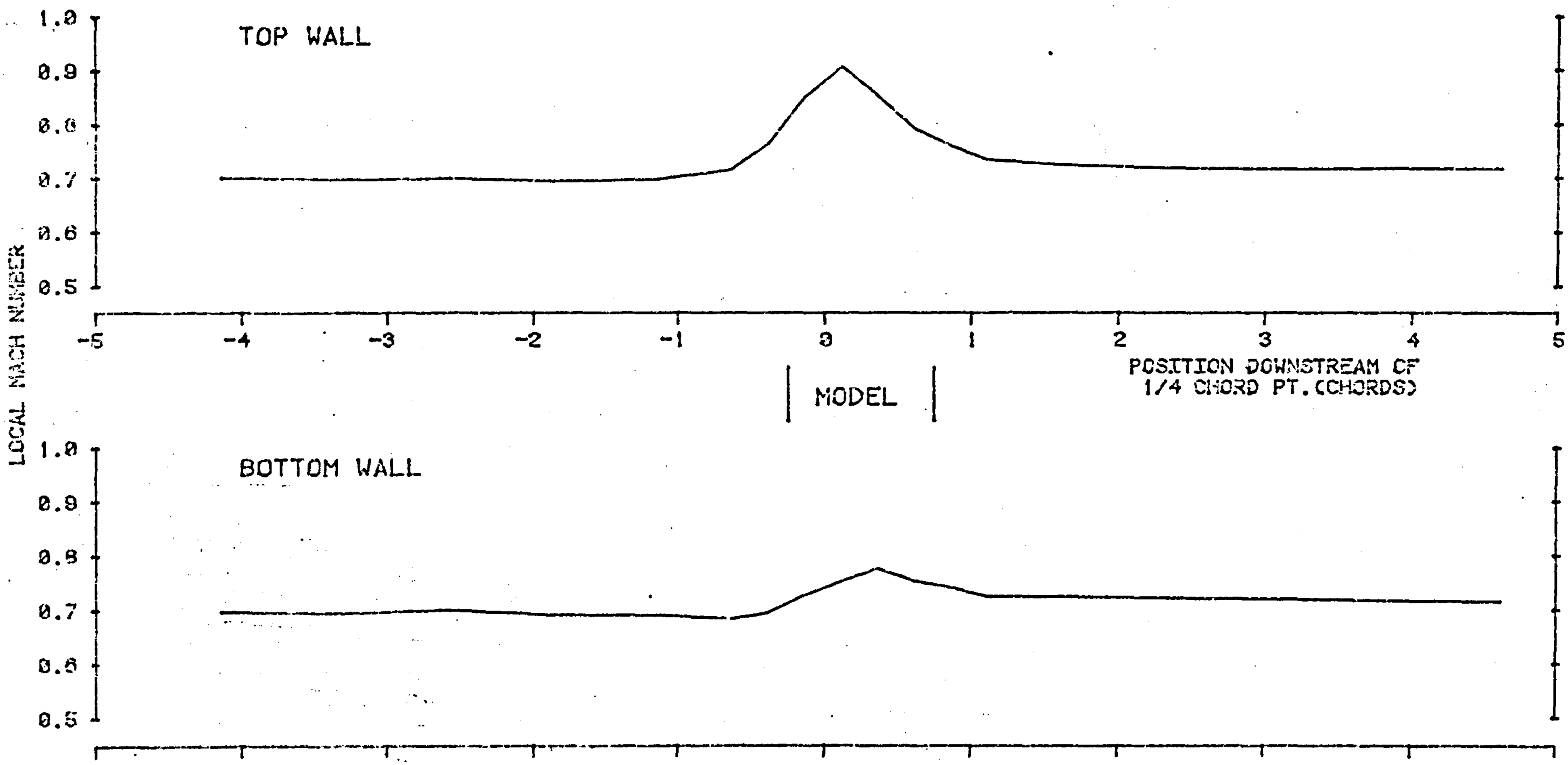


BOTH WALLS SET "STRAIGHT"

FIGURE 2.2

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 56 ALPHA 3.0 MACH NO 0.697

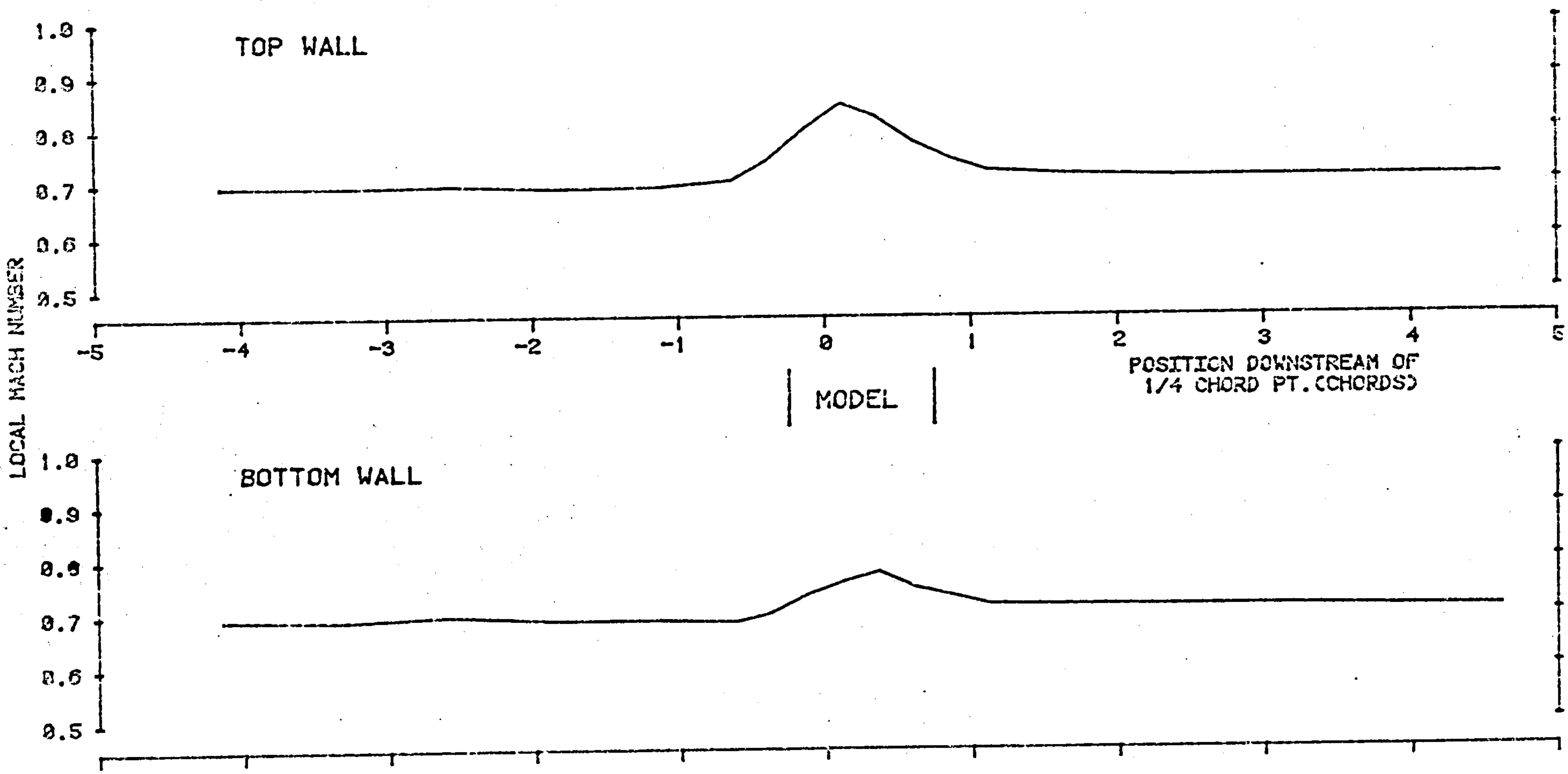


BOTH WALLS SET "STRAIGHT"

FIGURE 2.3

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
55	2.0	0.693

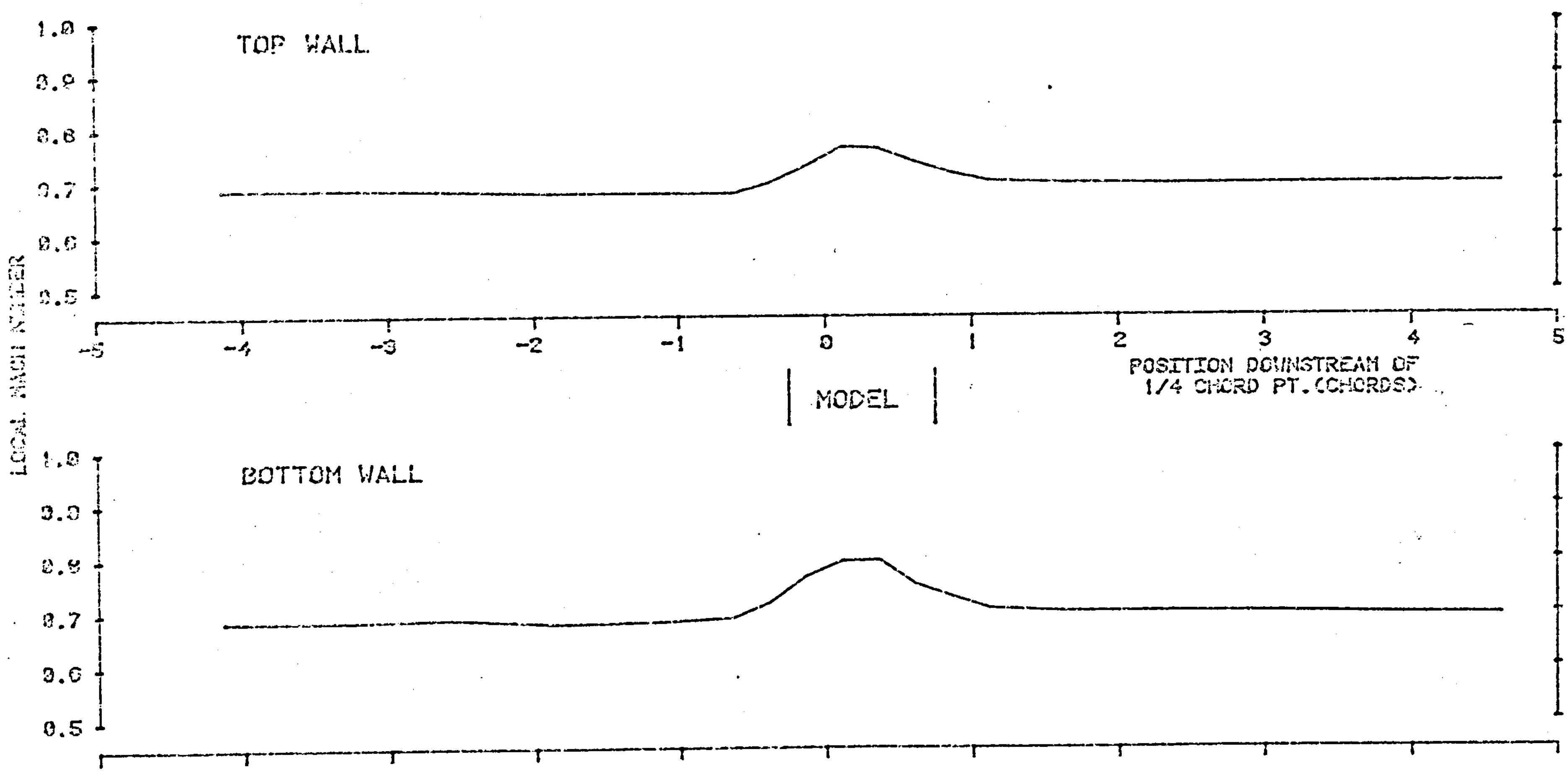


BOTH WALLS SET "STRAIGHT"

FIGURE 2.4

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
54	0.0	0.683

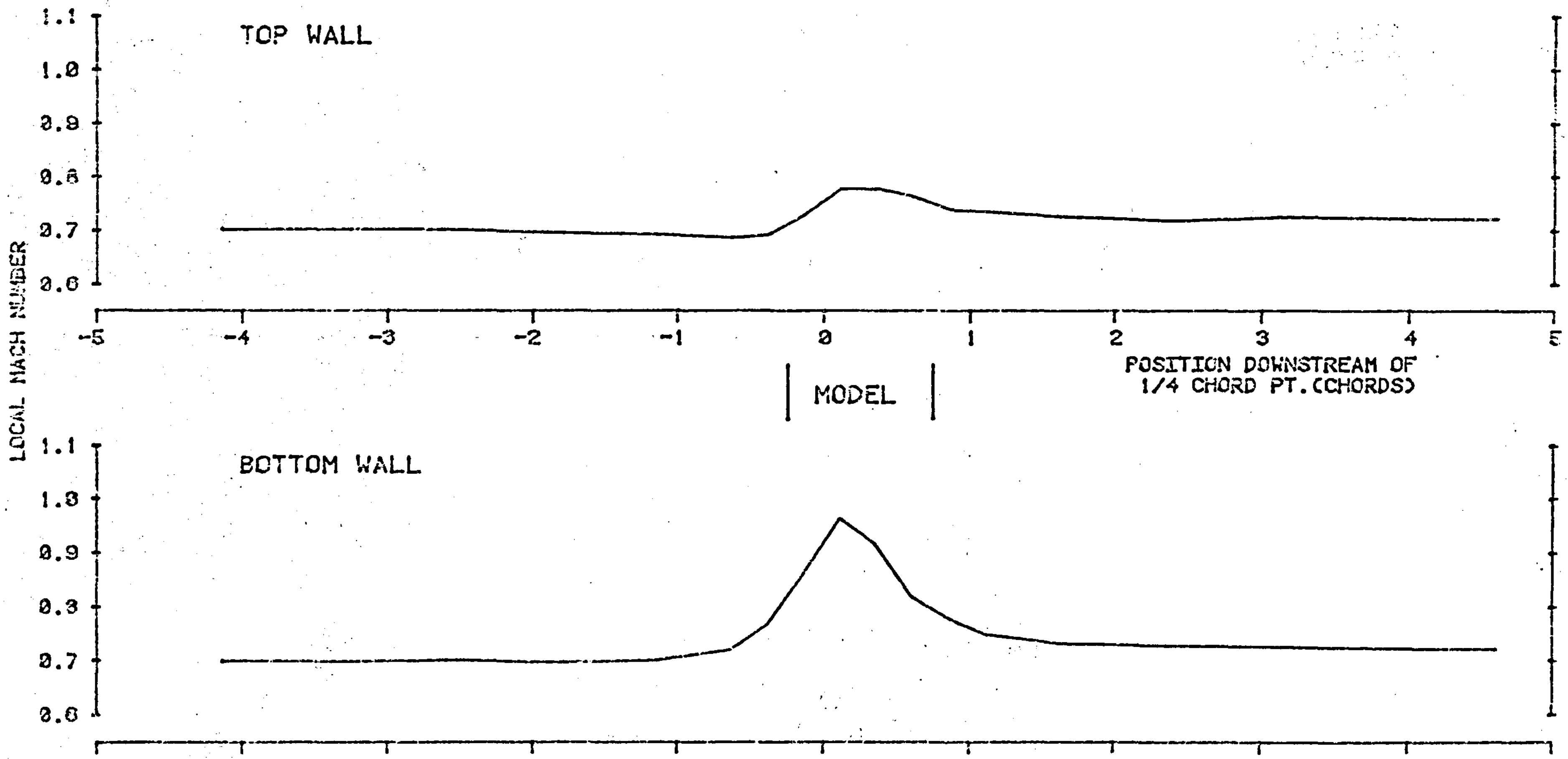


BOTH WALLS SET "STRAIGHT"

FIGURE 2.5

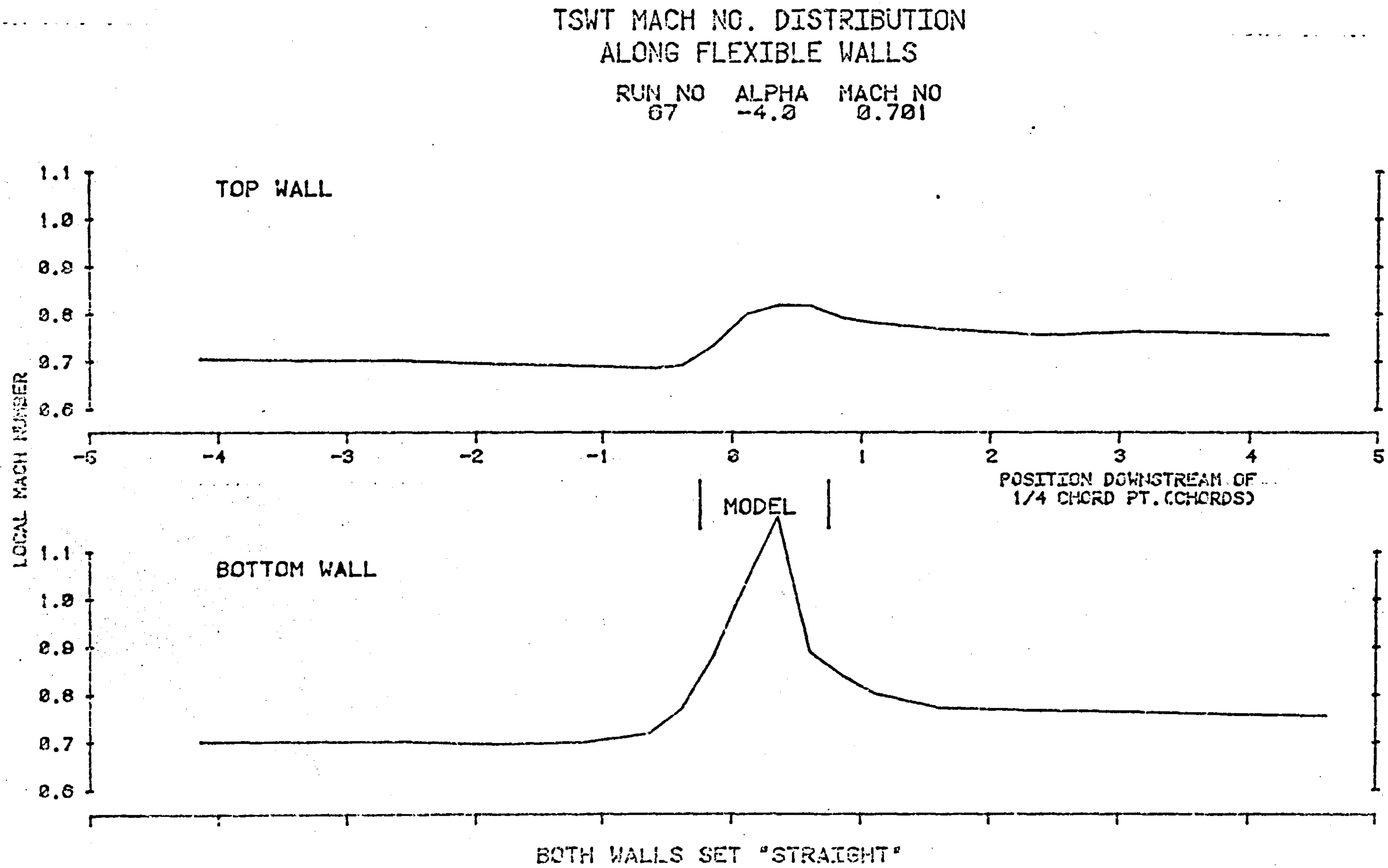
TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
63	-2.0	0.701



BOTH WALLS SET "STRAIGHT"

FIGURE 2.6



TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 67
ALPHA -4.0
MACH NO 0.701
TOP WALL
BOTTOM WALL
LOCAL MACH NUMBER
1.1
1.0
0.9
0.8
0.7
0.6
-5
-4
-3
-2
-1
0
1
2
3
4
5
POSITION DOWNSTREAM OF
1/4 CHORD PT. (CHORDS)
MODEL
BOTH WALLS SET "STRAIGHT"

FIGURE 2.7

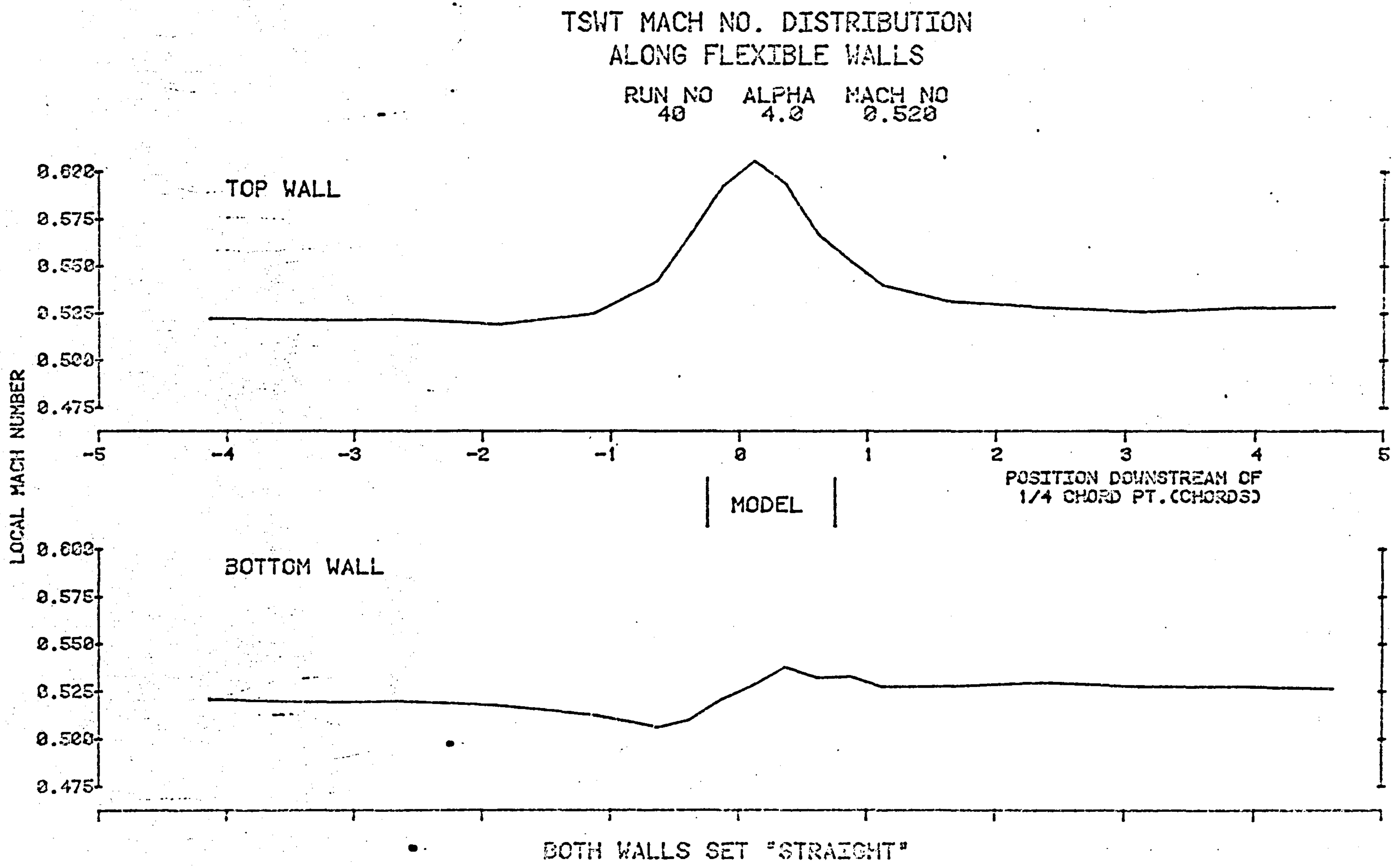
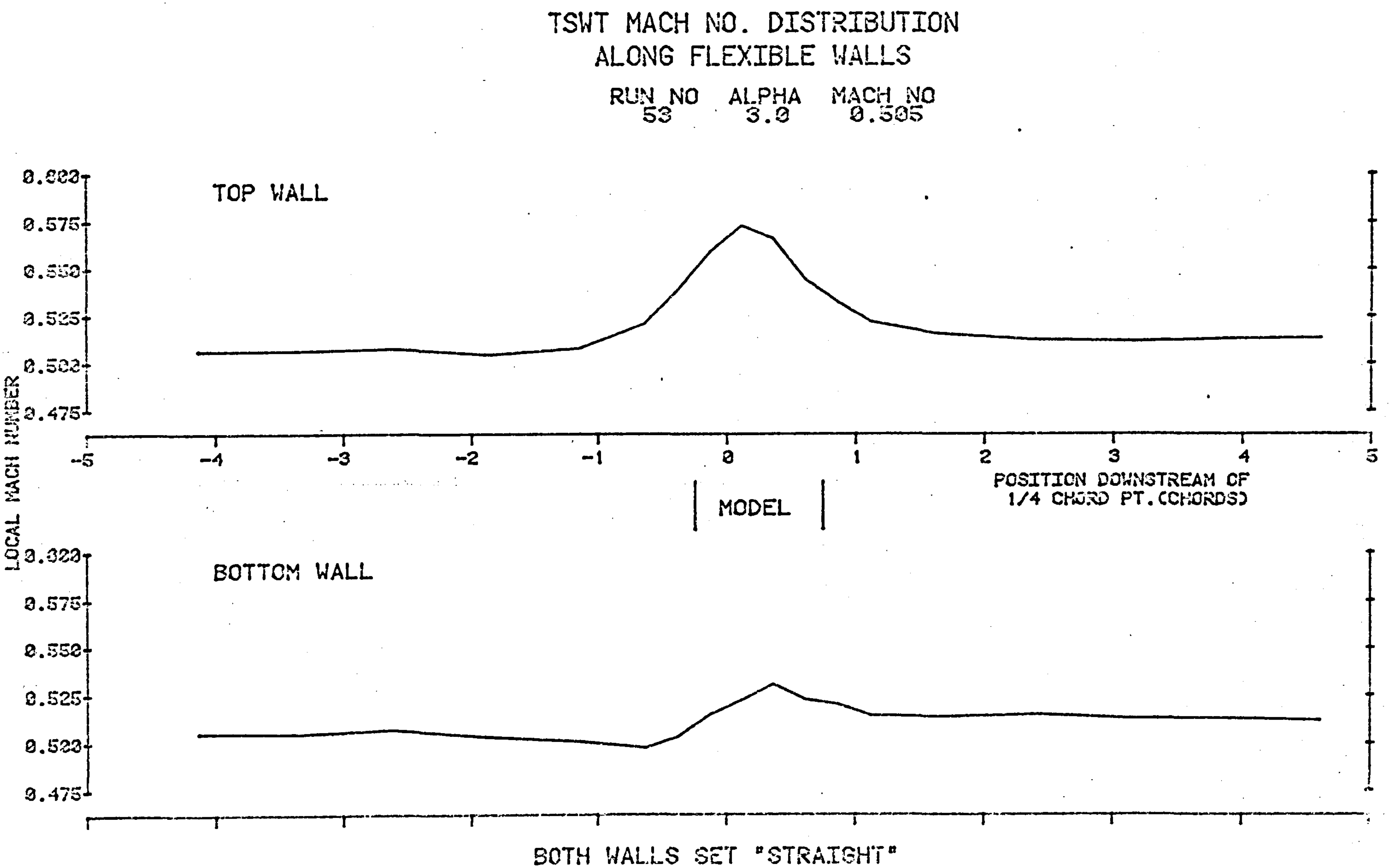

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 40
ALPHA 4.0
MACH NO 0.520
TOP WALL
BOTTOM WALL
LOCAL MACH NUMBER
POSITION DOWNSTREAM OF 1/4 CHORD PT.(CHORDS)
MODEL
BOTH WALLS SET "STRAIGHT"

FIGURE 2.8

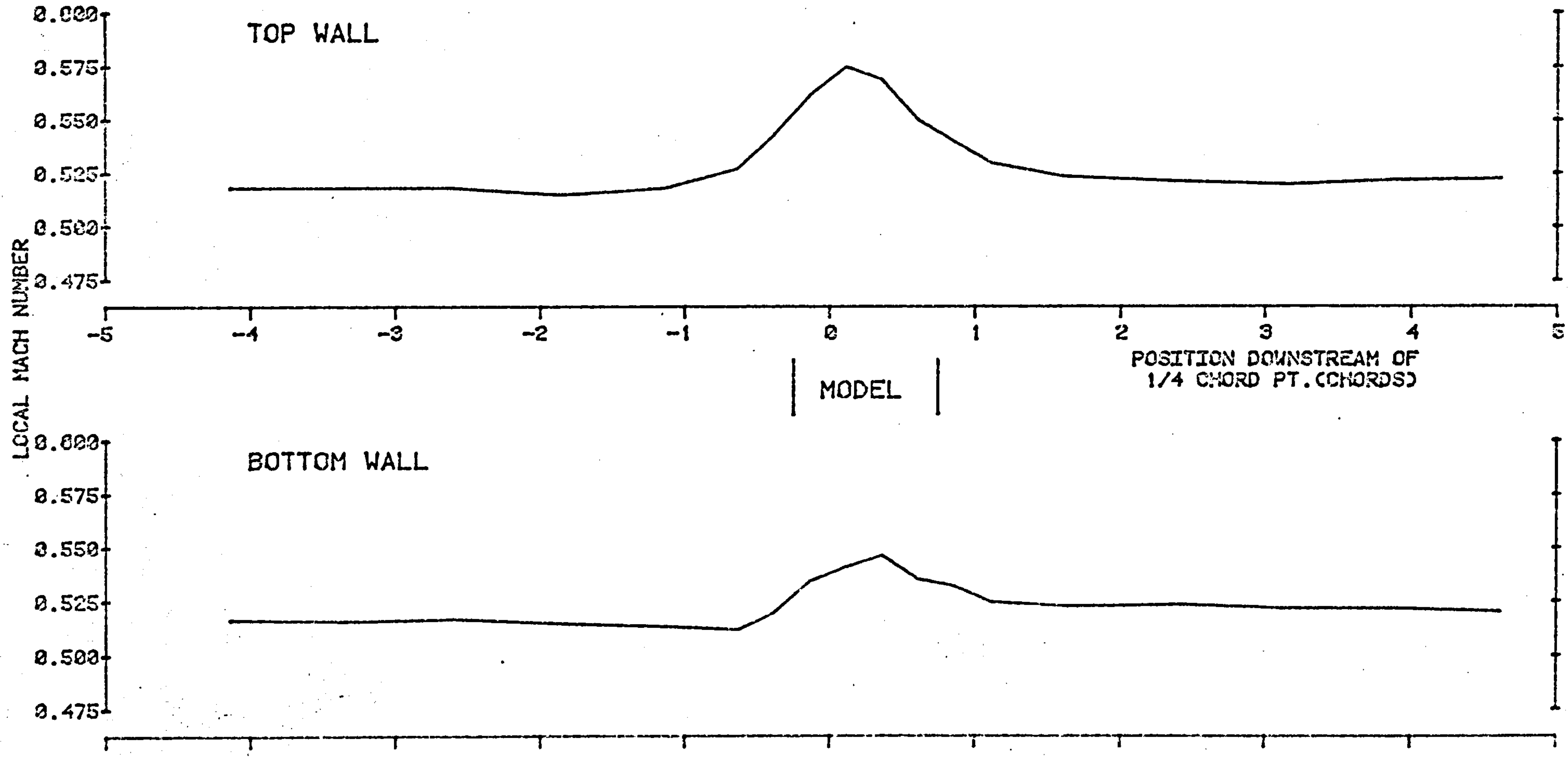


TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 53
ALPHA 3.0
MACH NO 0.505
TOP WALL
BOTTOM WALL
LOCAL MACH NUMBER
0.600
0.575
0.550
0.525
0.500
0.475
-5
-4
-3
-2
-1
0
1
2
3
4
5
MODEL
POSITION DOWNSTREAM OF
1/4 CHORD PT. (CHORDS)
BOTH WALLS SET "STRAIGHT"

FIGURE 2.9

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
39	2.0	0.516

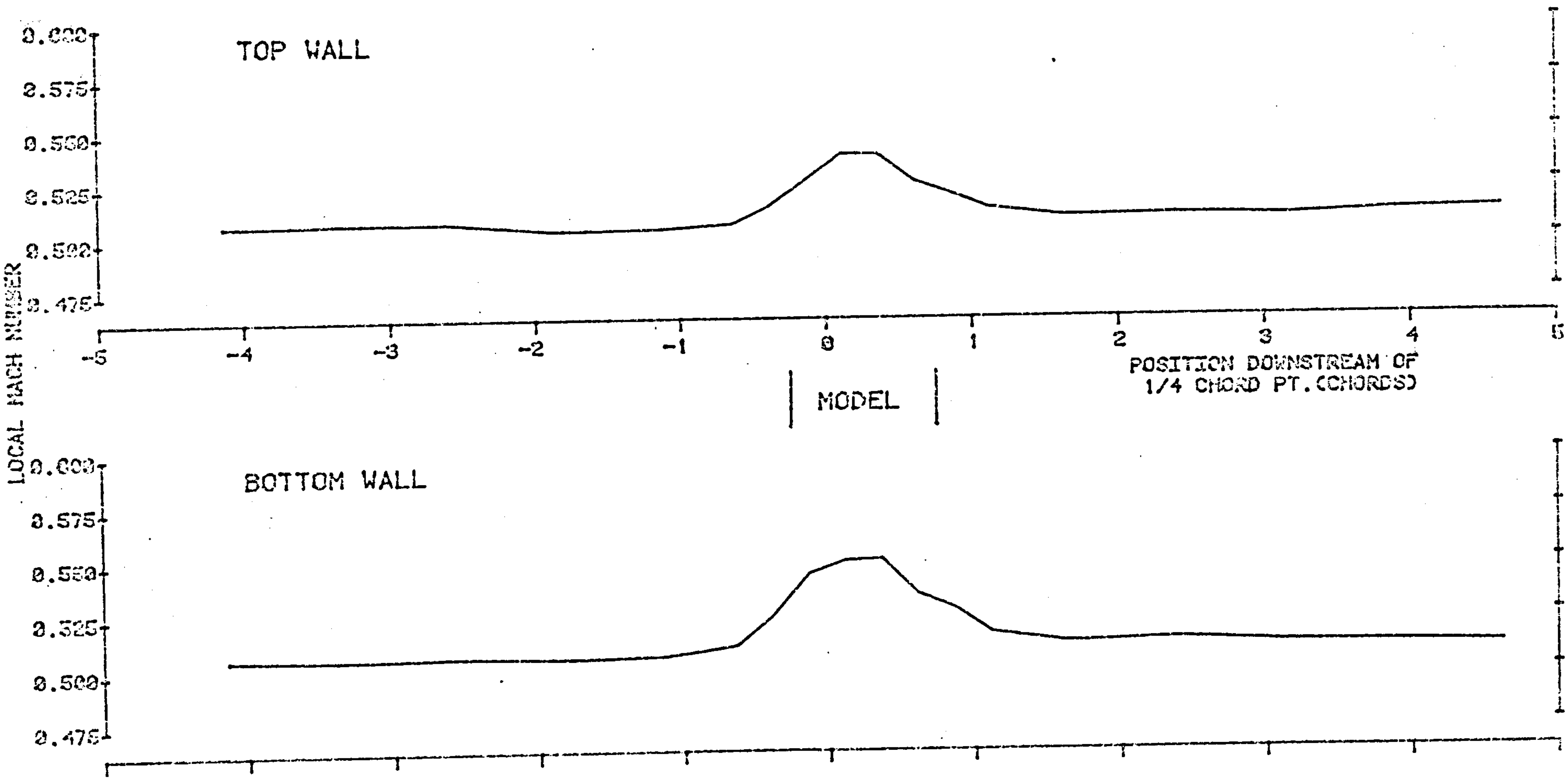


BOTH WALLS SET "STRAIGHT"

FIGURE 2.10

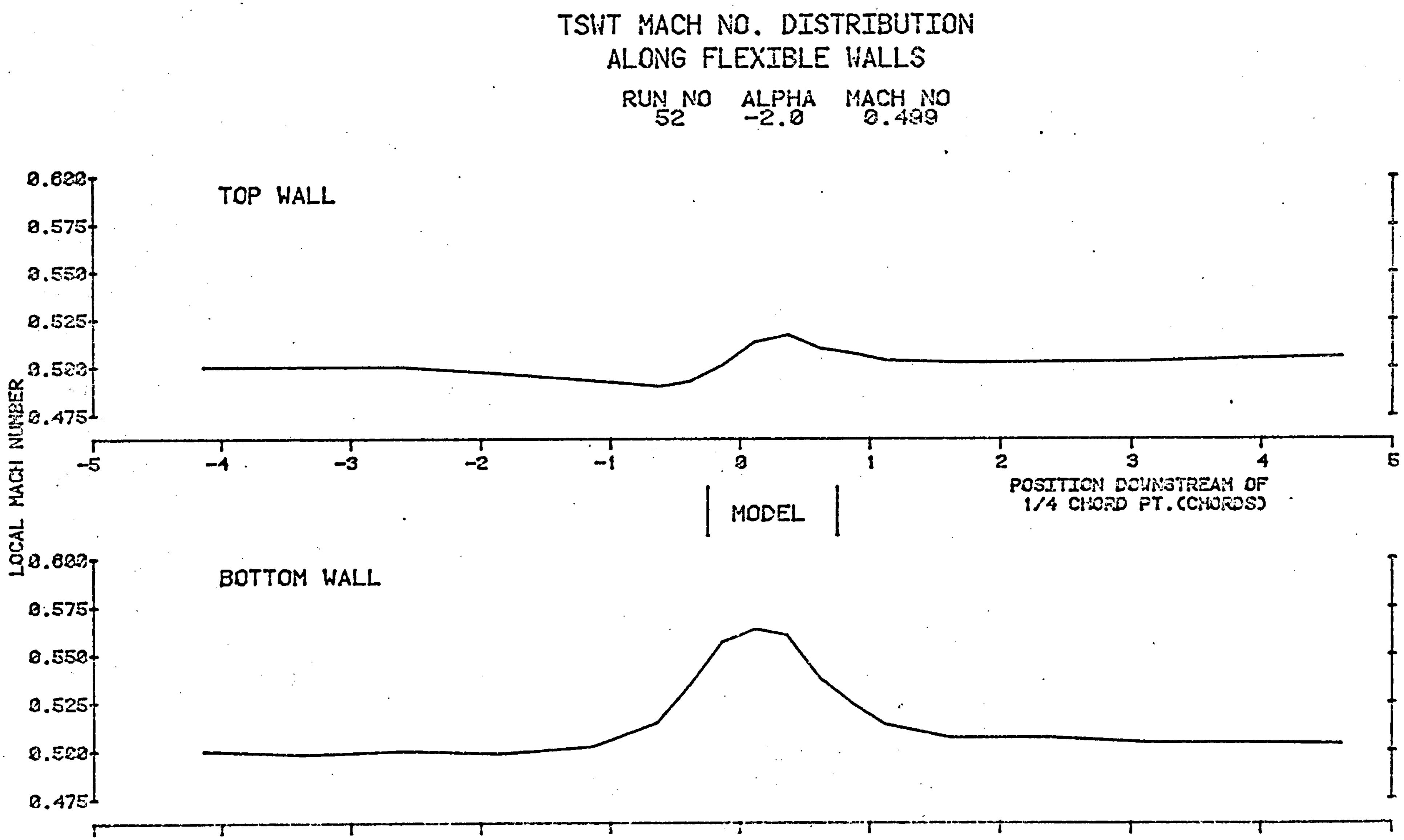
TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 36 ALPHA 0.0 MACH NO 0.505



BOTH WALLS SET "STRAIGHT"

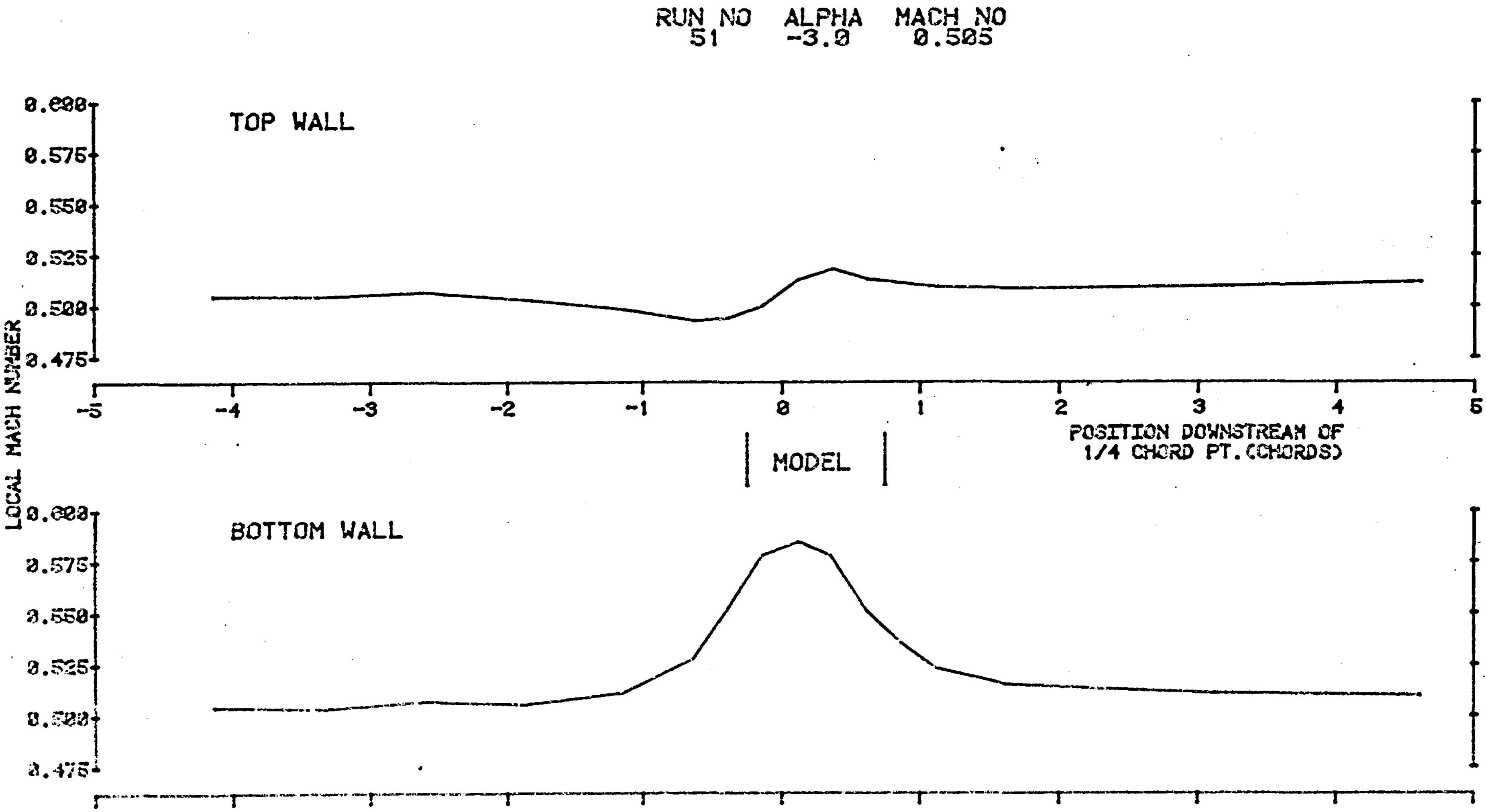
FIGURE 2.11



BOTH WALLS SET "STRAIGHT"

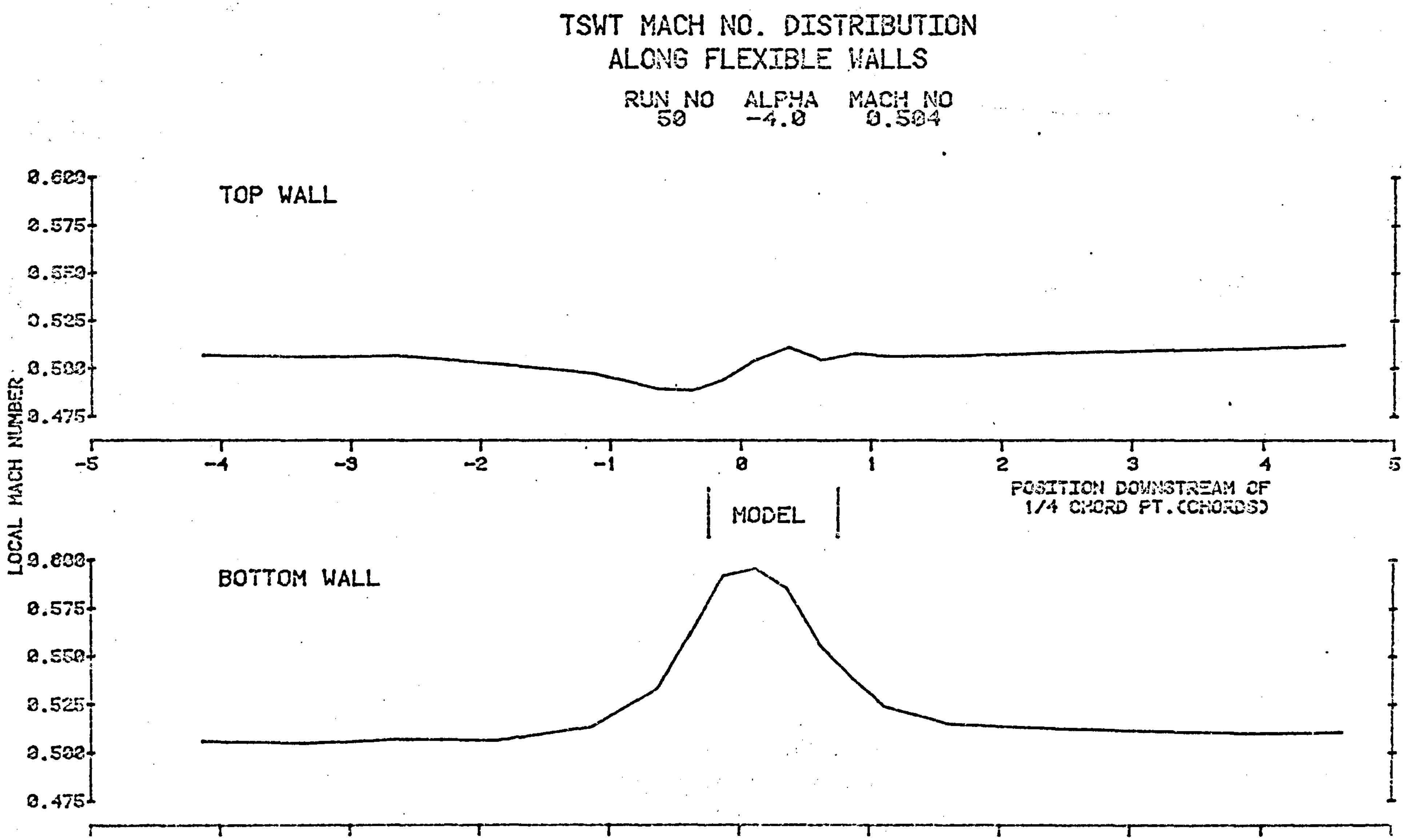
FIGURE 2.12

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS



BOTH WALLS SET "STRAIGHT"

FIGURE 2.13

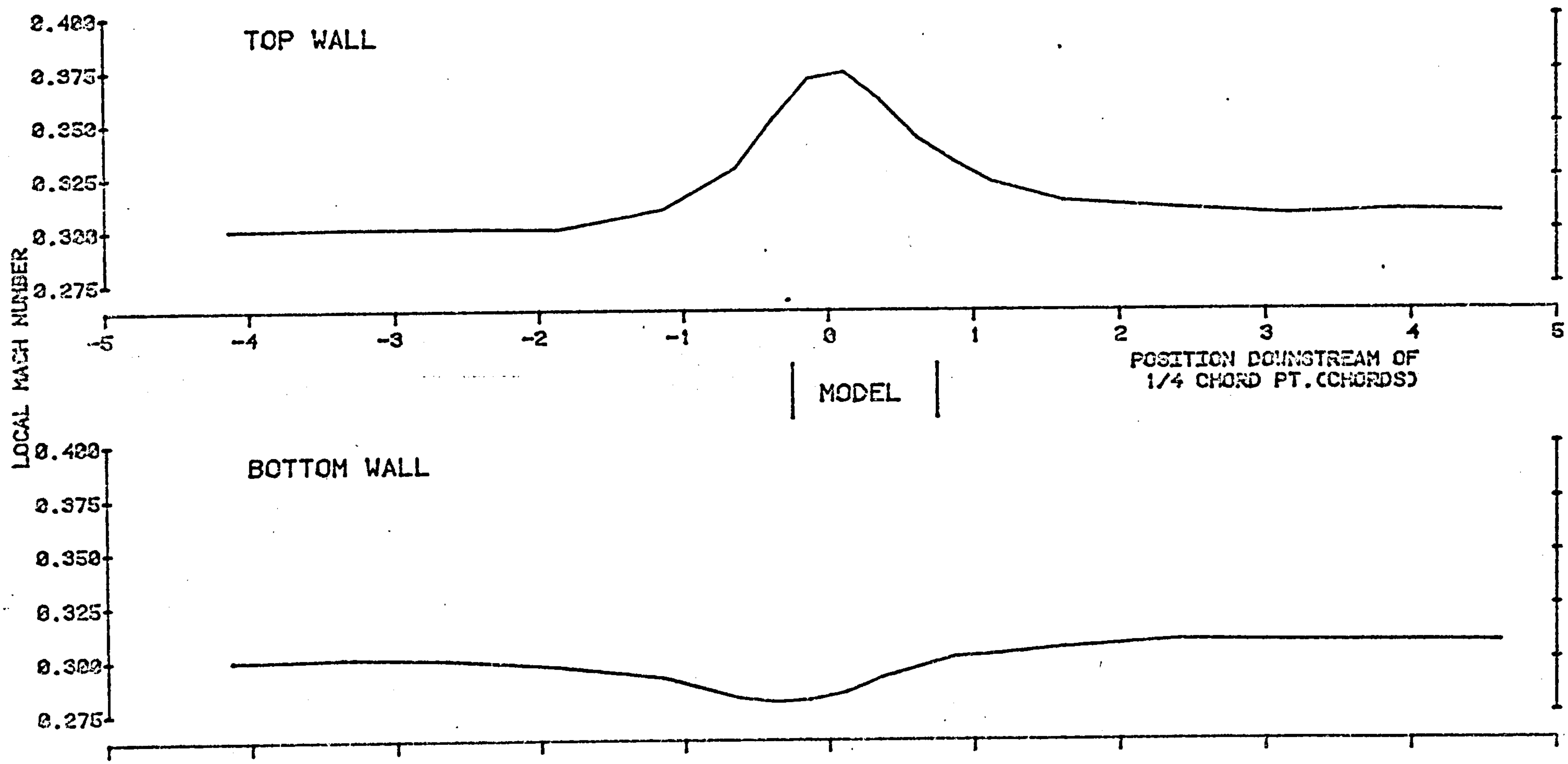


BOTH WALLS SET "STRAIGHT"

FIGURE 2.14

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
44	10.0	0.301



BOTH WALLS SET "STRAIGHT"

FIGURE 2.15

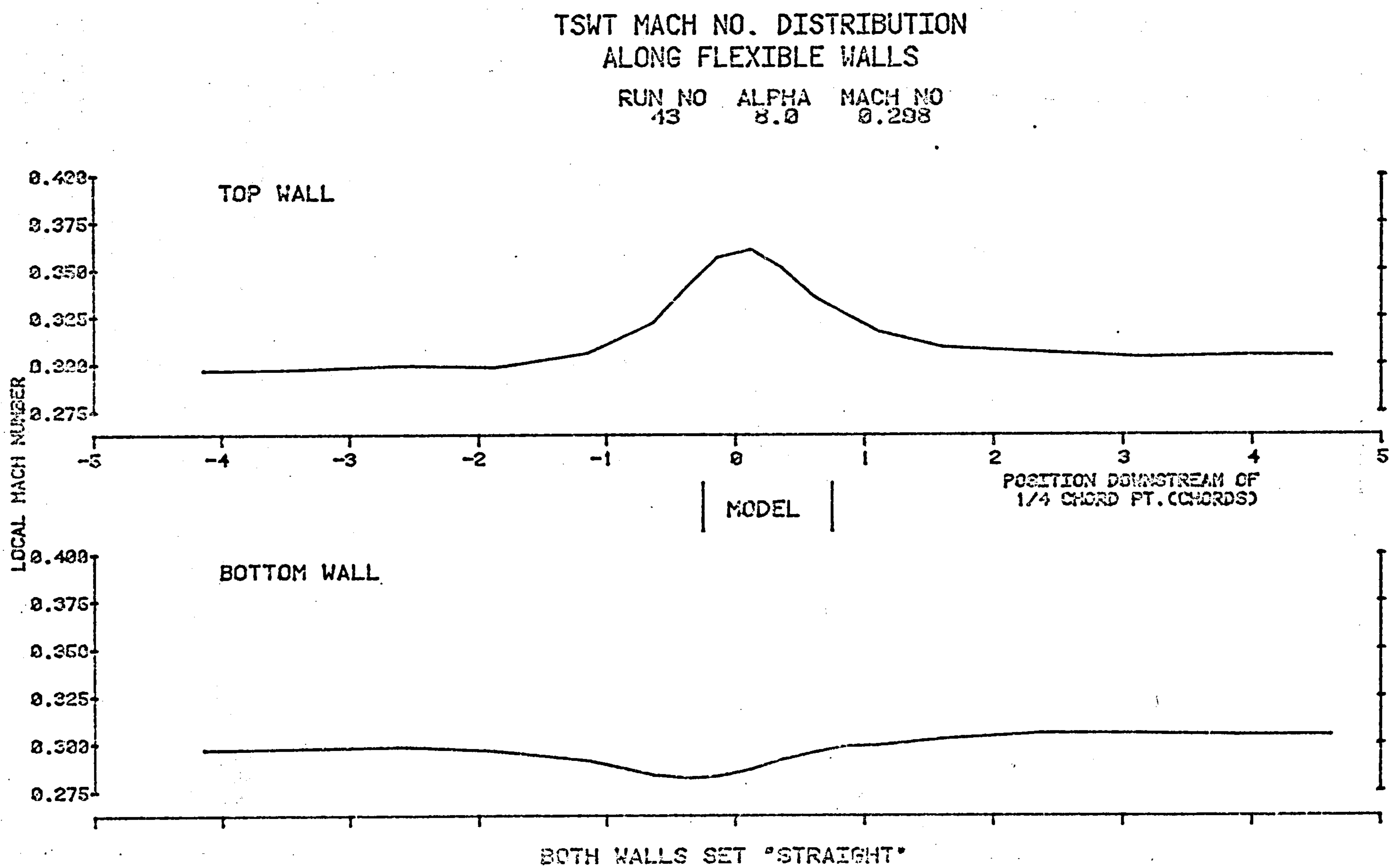
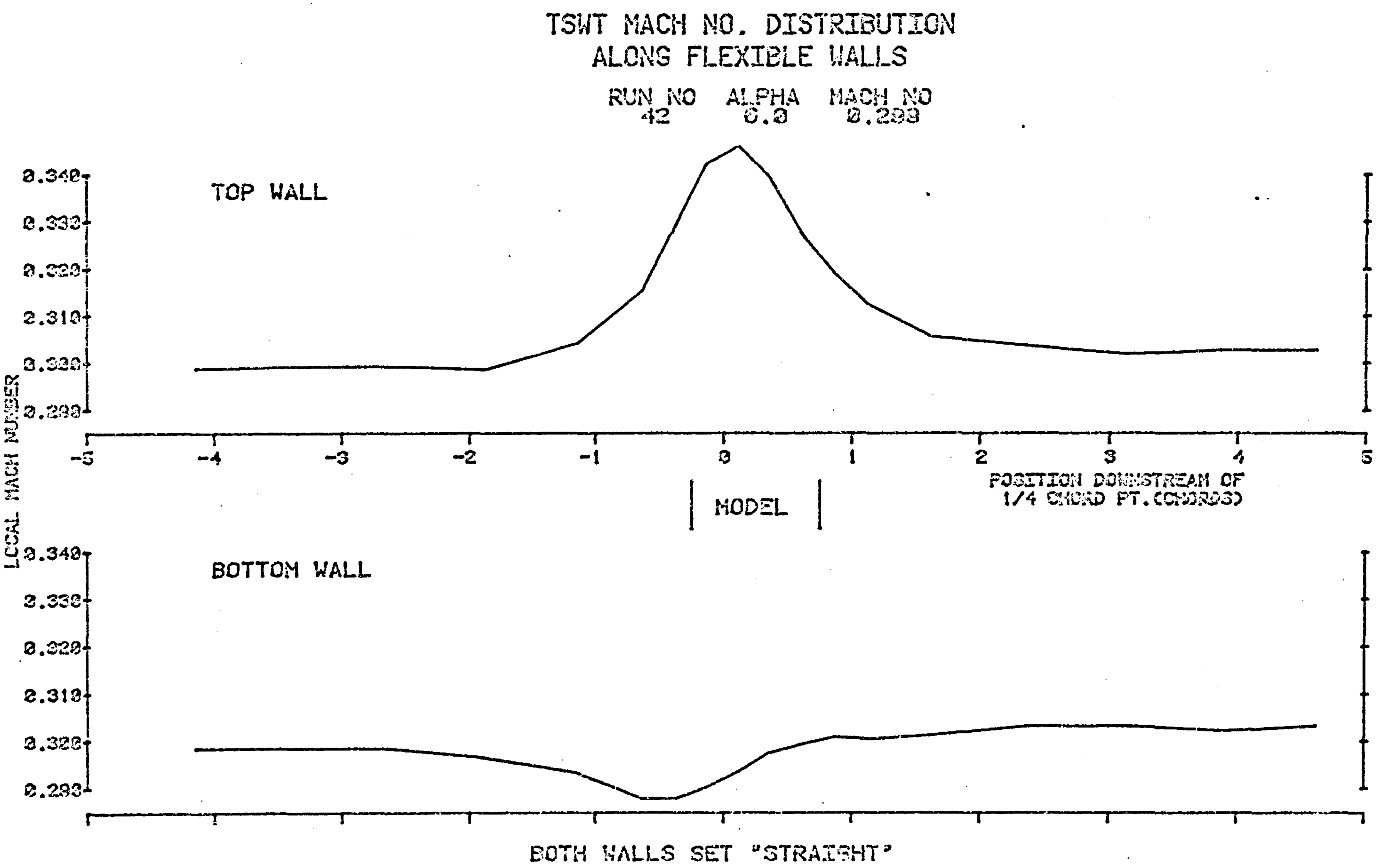

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 43
ALPHA 8.0
MACH NO 0.298
TOP WALL
BOTTOM WALL
LOCAL MACH NUMBER
0.400
0.375
0.350
0.325
0.300
0.275
-5
-4
-3
-2
-1
0
1
2
3
4
5
POSITION DOWNSTREAM OF 1/4 CHORD PT. (CHORDS)
MODEL
BOTH WALLS SET "STRAIGHT"

FIGURE 2.16

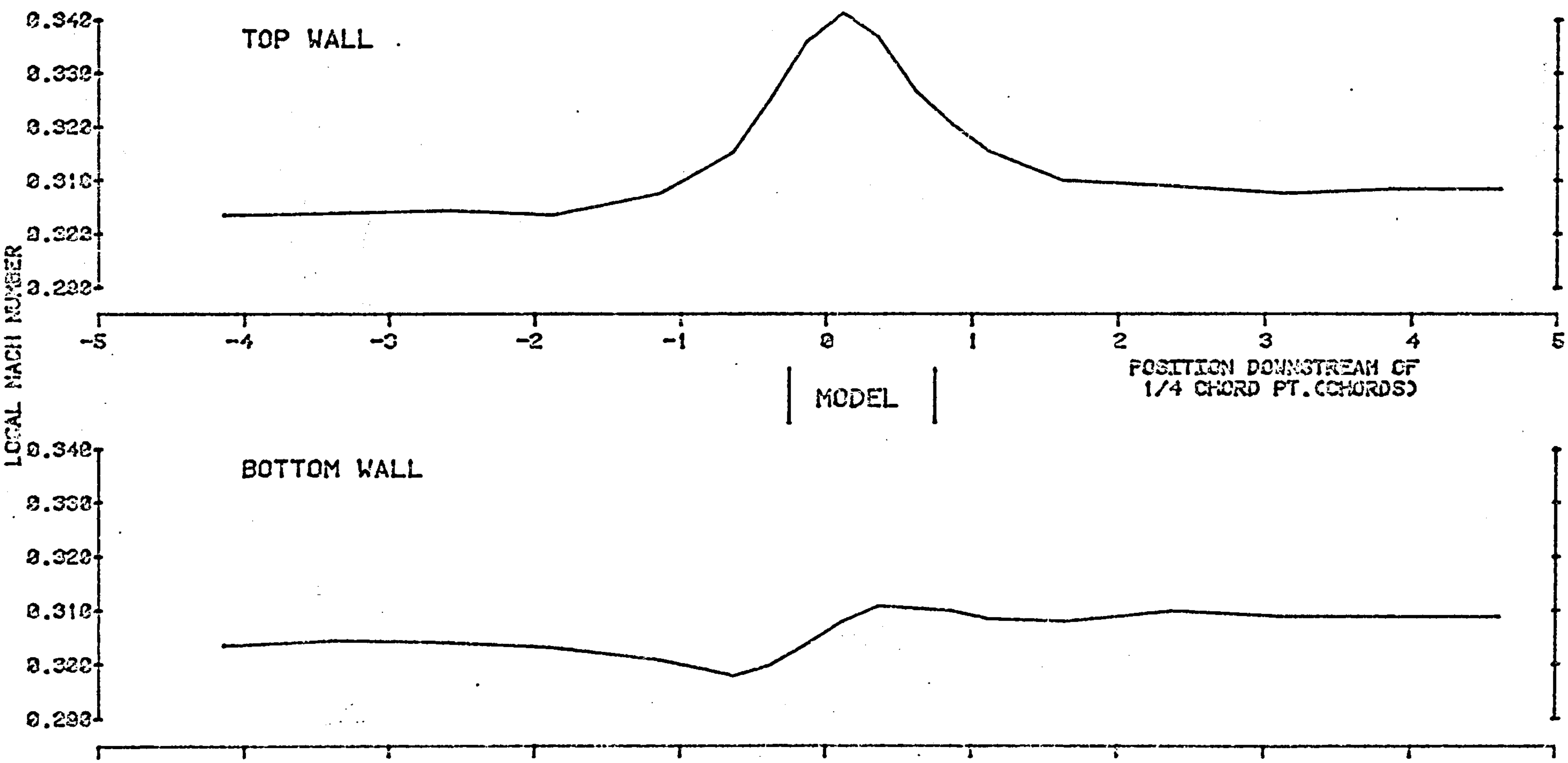


- 74 -

FIGURE 2.17

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
42	4.0	0.304

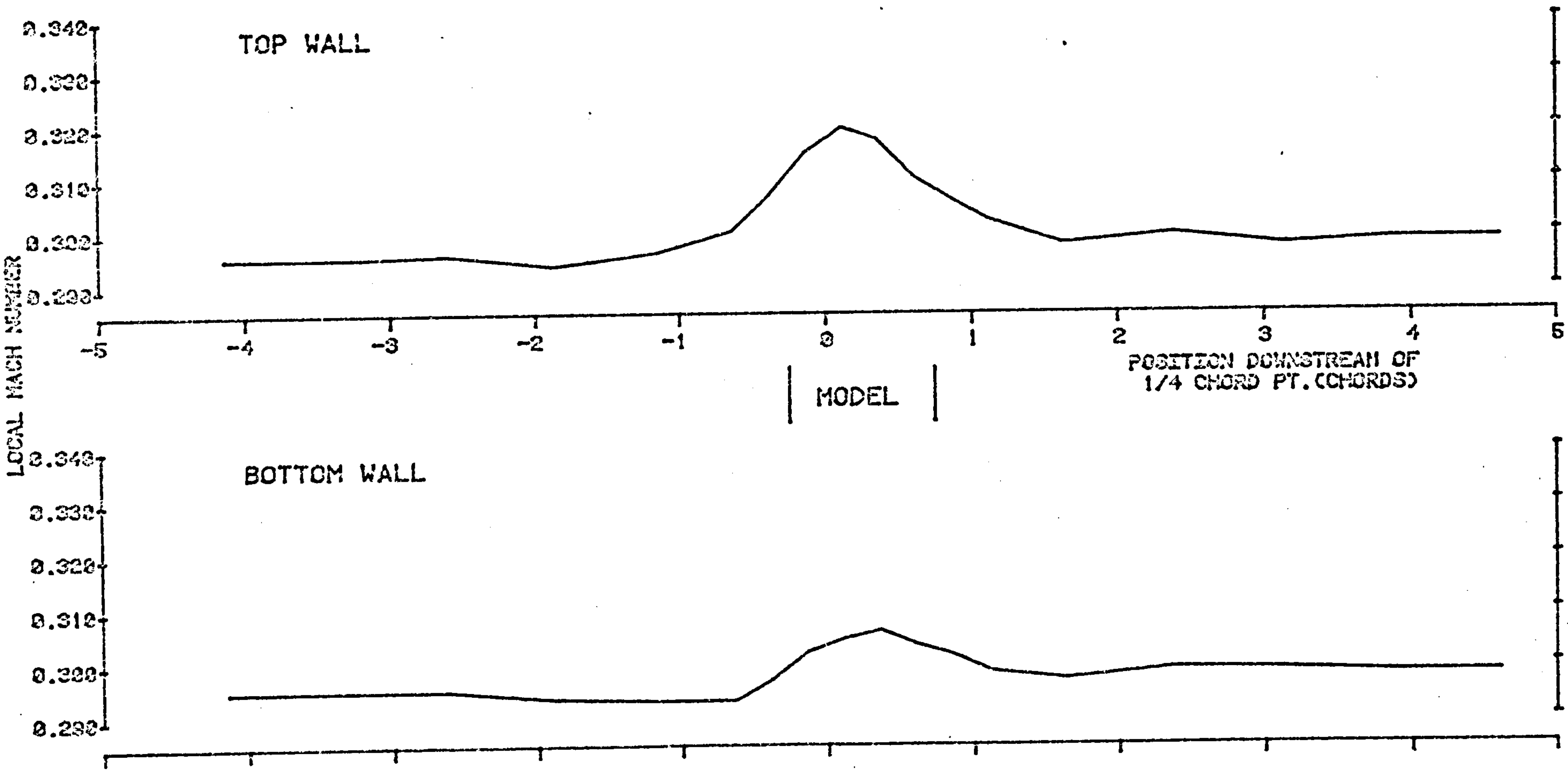


BOTH WALLS SET "STRAIGHT"

FIGURE 2.18

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 41 ALPHA 2.0 MACH NO 0.296

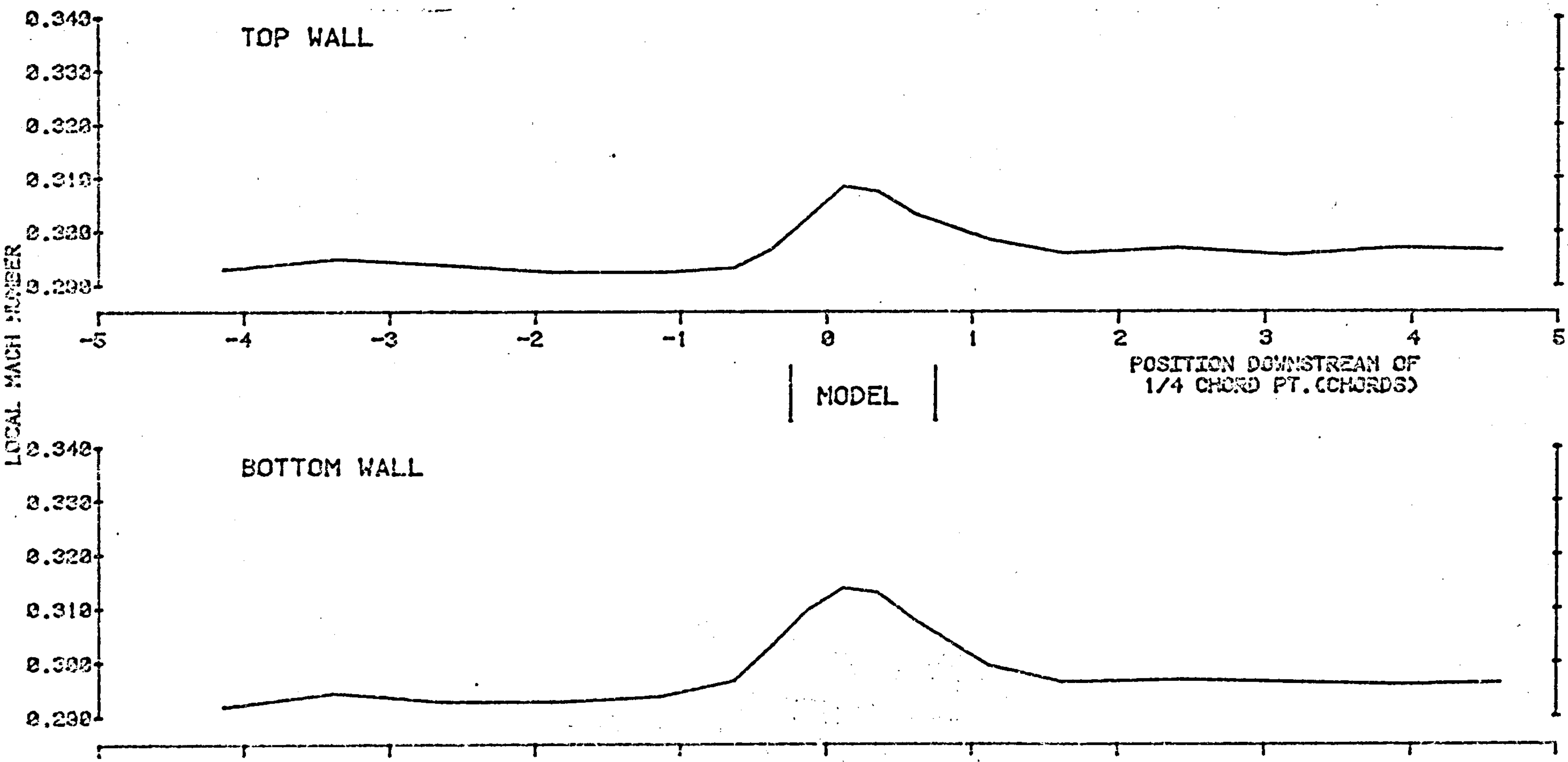


BOTH WALLS SET "STRAIGHT"

FIGURE 2.19

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO	ALPHA	MACH NO
40	0.0	0.293



BOTH WALLS SET "STRAIGHT"

FIGURE 2.20

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

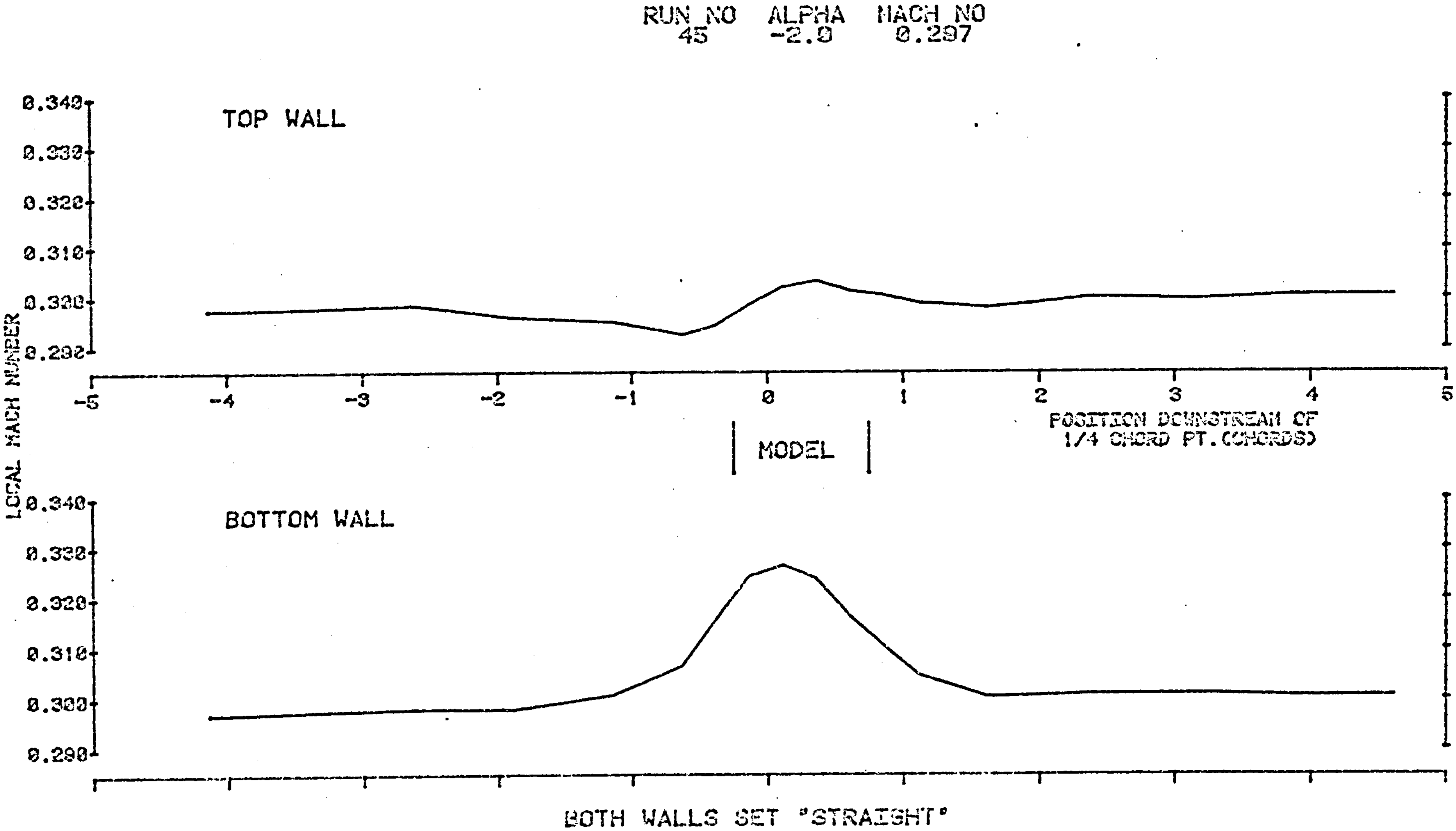


FIGURE 2.21

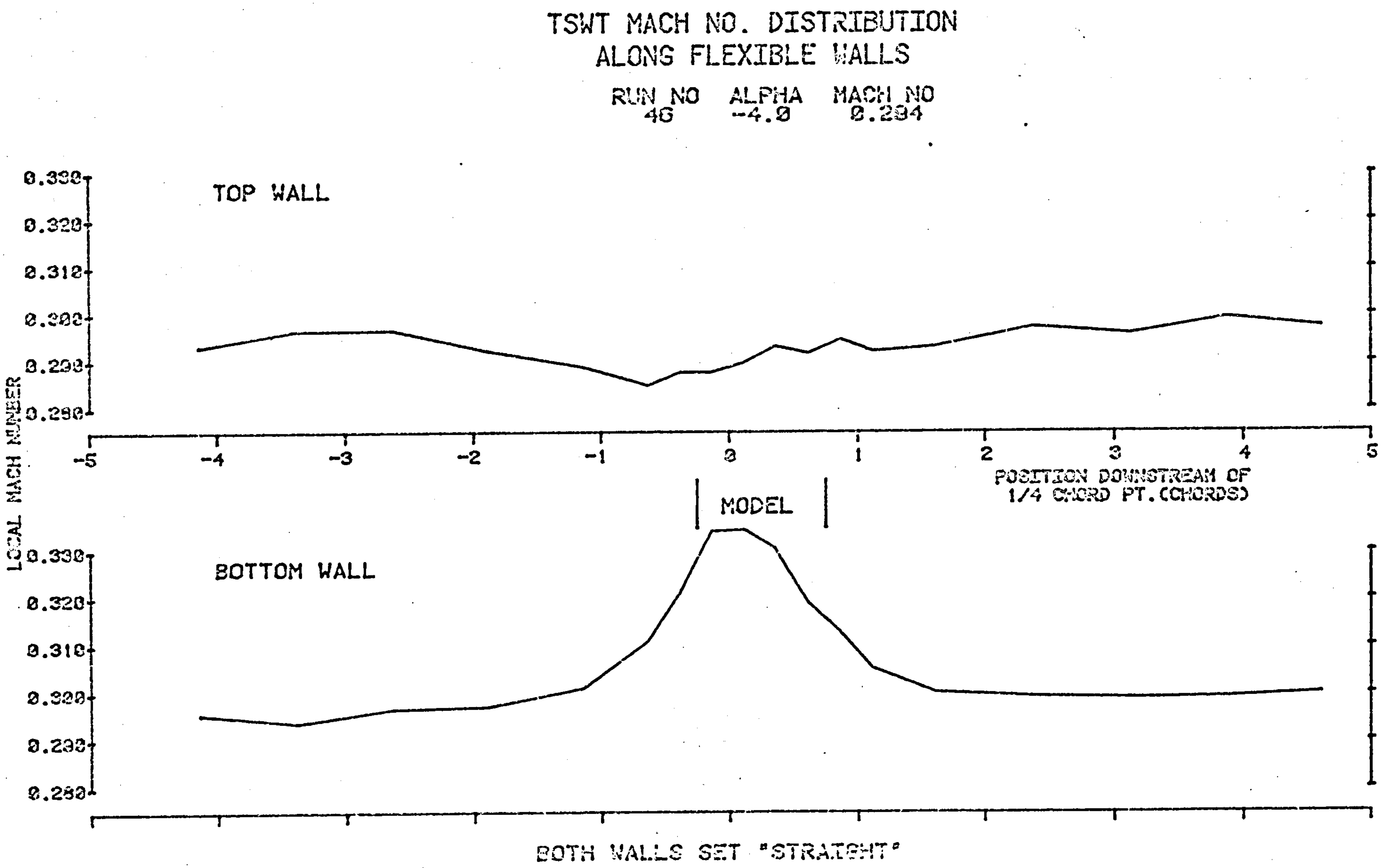
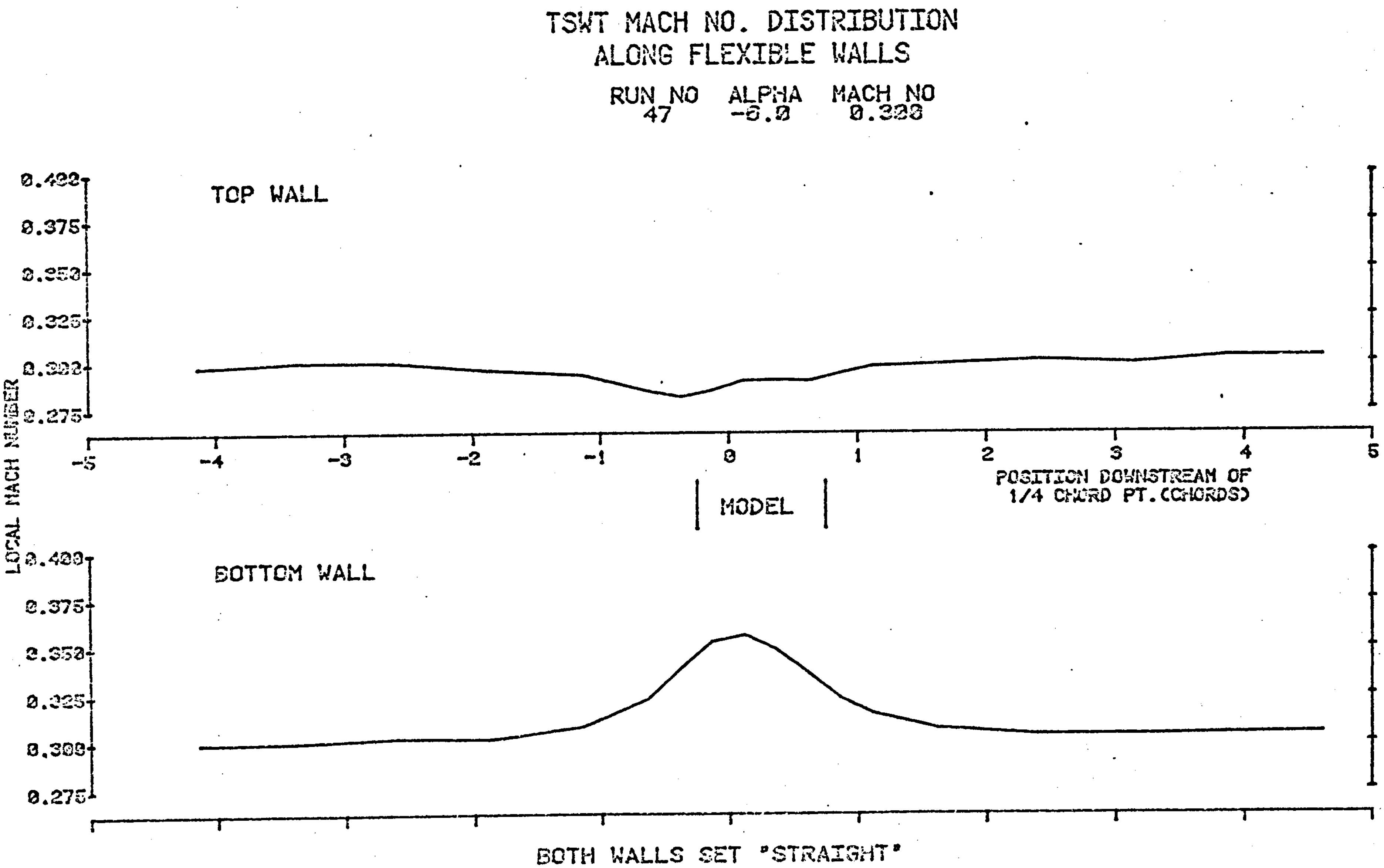

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 46
ALPHA -4.0
MACH NO 0.294
TOP WALL
0.330
0.320
0.310
0.300
0.290
0.280
LOCAL MACH NUMBER
-5
-4
-3
-2
-1
0
1
2
3
4
5
POSITION DOWNSTREAM OF
1/4 CHORD PT.(CHORDS)
MODEL
BOTTOM WALL
0.330
0.320
0.310
0.300
0.290
0.280
BOTH WALLS SET "STRAIGHT"

FIGURE 2.22

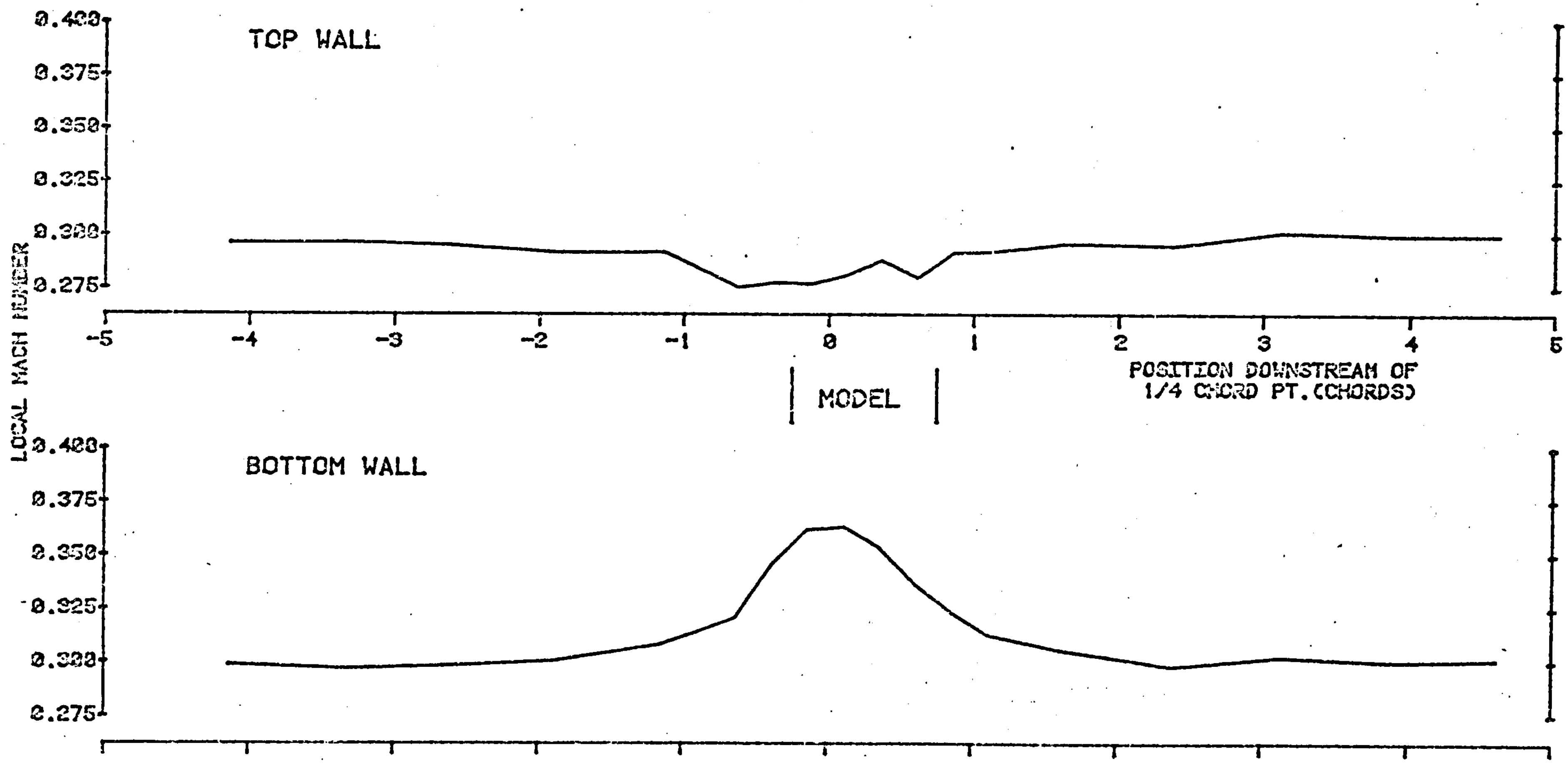


TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS
RUN NO 47
ALPHA -6.0
MACH NO 0.300
TOP WALL
BOTTOM WALL
LOCAL MACH NUMBER
0.400
0.375
0.350
0.325
0.300
0.275
-5
-4
-3
-2
-1
0
1
2
3
4
5
POSITION DOWNSTREAM OF 1/4 CHORD PT. (CHORDS)
MODEL
BOTH WALLS SET "STRAIGHT"

FIGURE 2.23

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 48 ALPHA -3.0 MACH NO 0.296

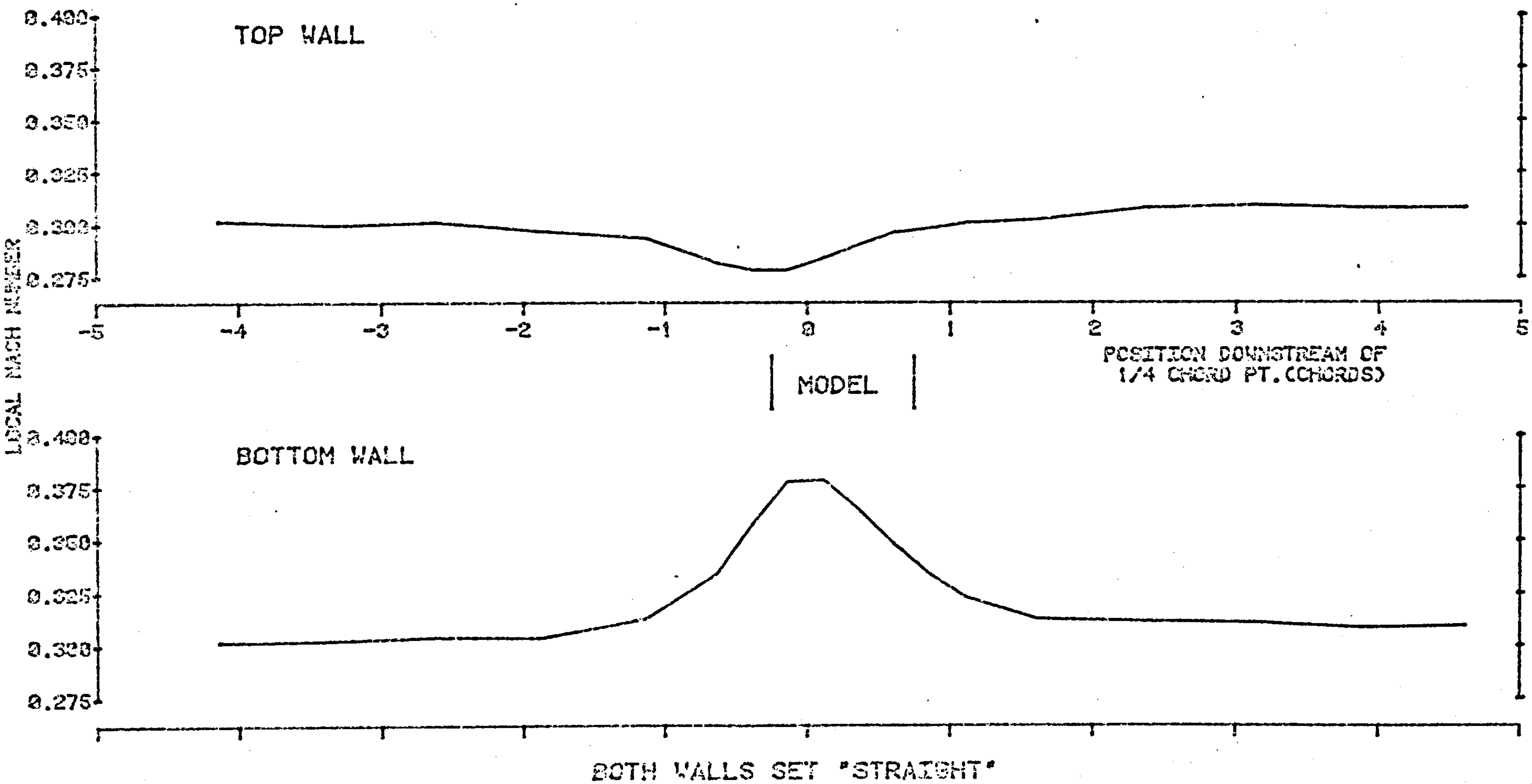


BOTH WALLS SET "STRAIGHT"

FIGURE 2.24

TSWT MACH NO. DISTRIBUTION
ALONG FLEXIBLE WALLS

RUN NO 49 ALPHA -10.0 MACH NO 0.301



NACA 0012-64 Section

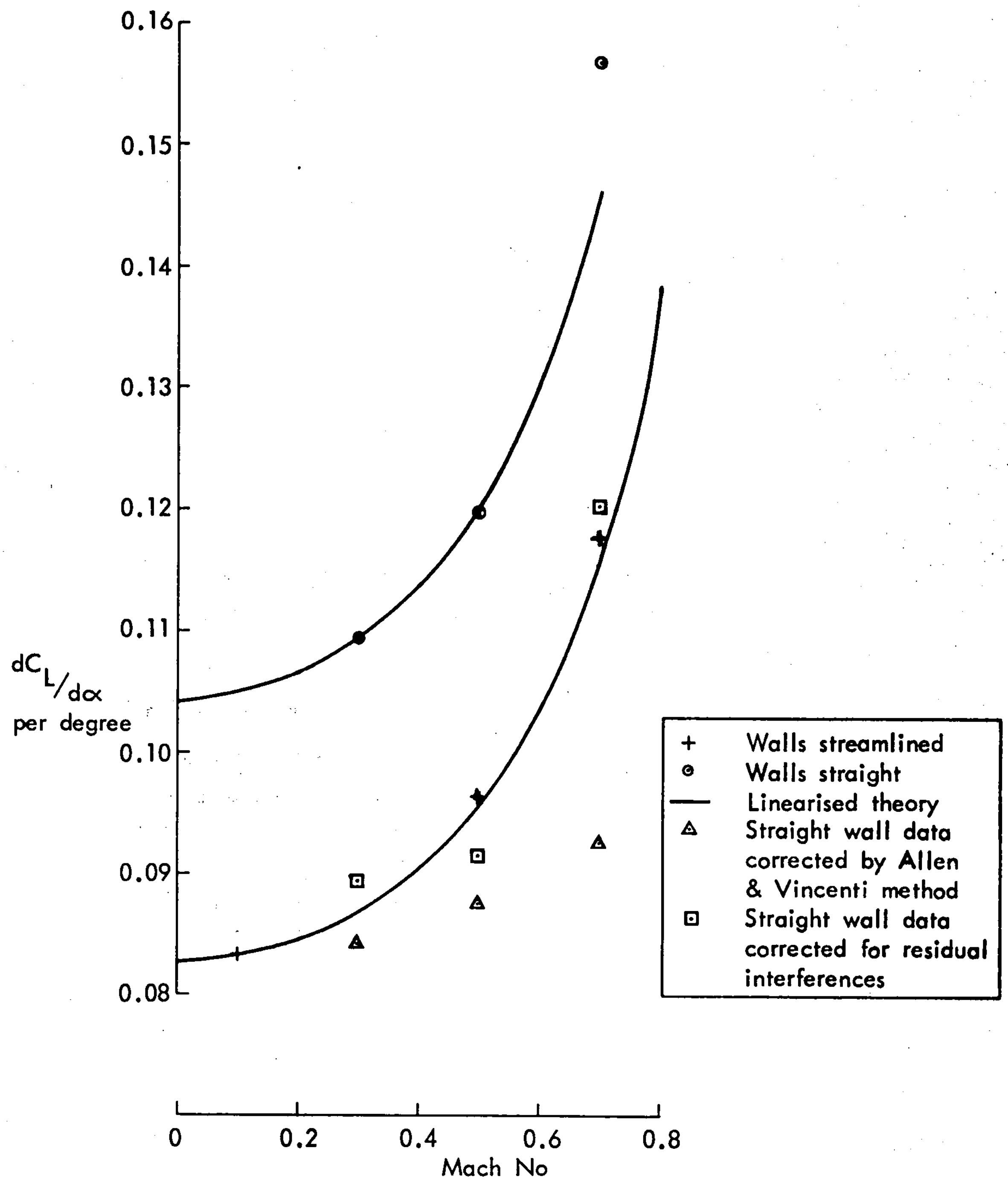


FIG. 3 SUMMARY OF MODEL DATA FROM FLEXIBLE WALLED WIND TUNNELS BELOW MACH 0.8

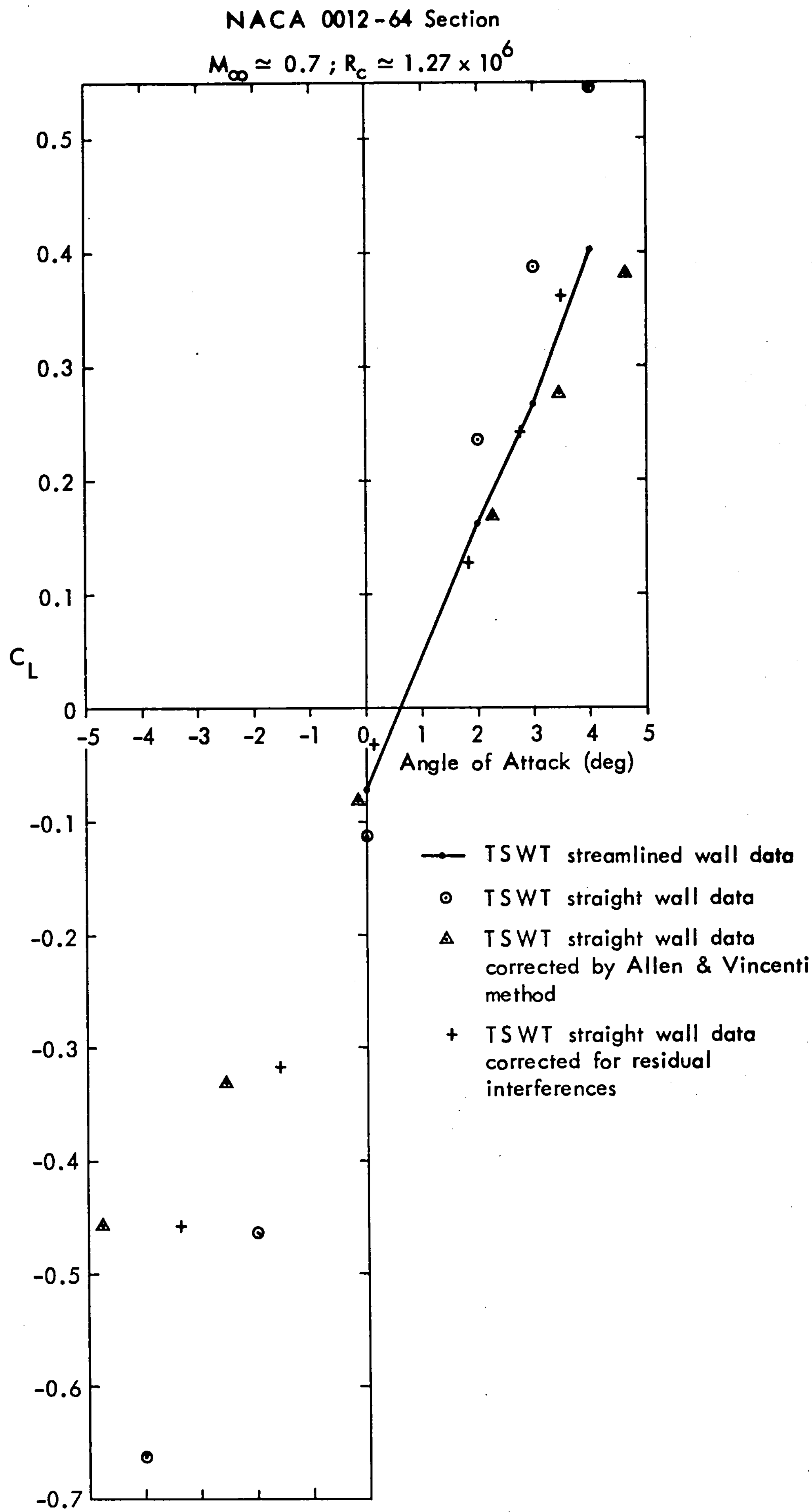


FIG. 4(a) LIFT CURVE SLOPES; $M_\infty \simeq 0.7$

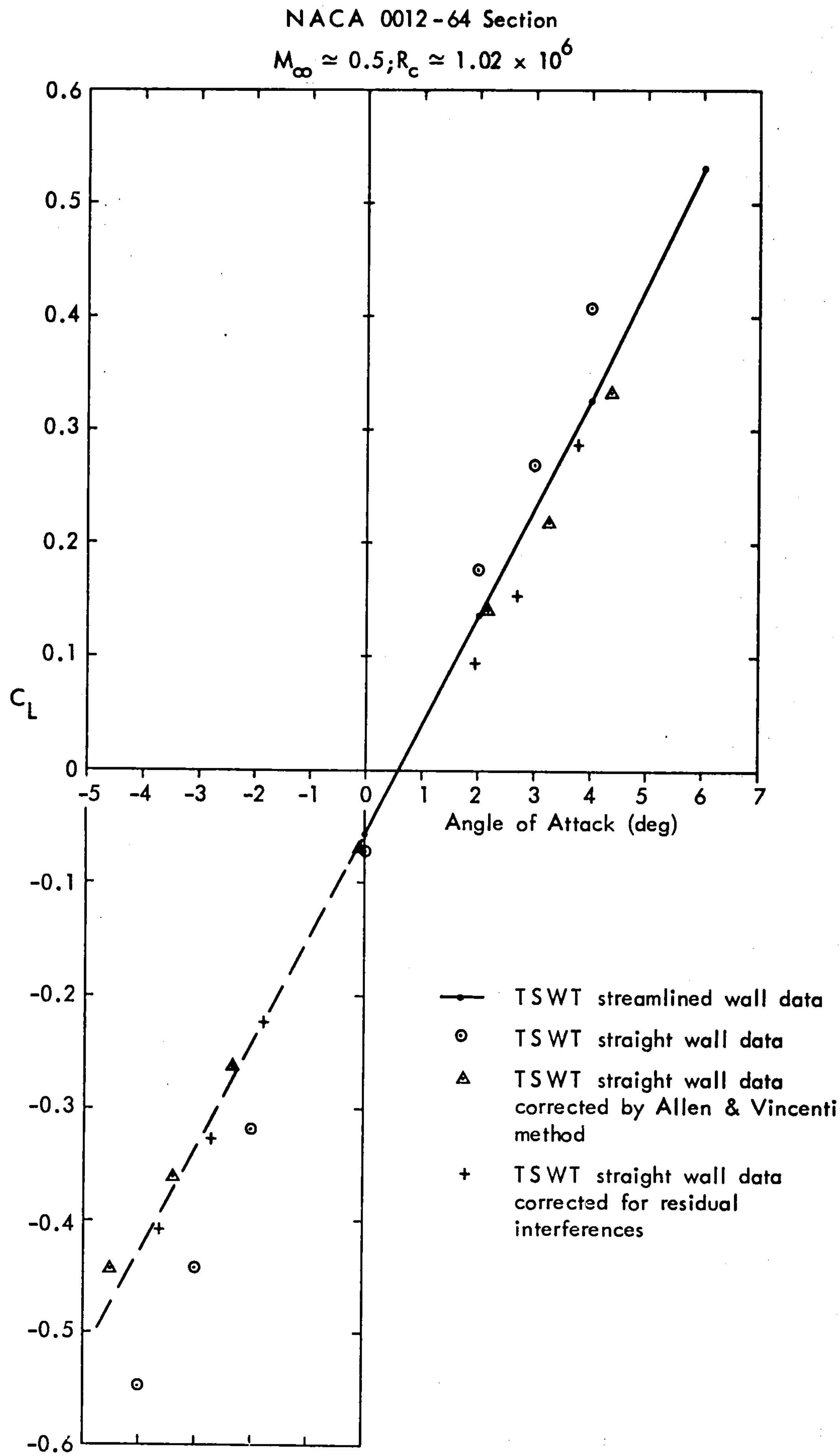


FIG. 4(b) LIFT CURVE SLOPES ; $M_\infty \simeq 0.5$

NACA 0012-64 Section

$M_\infty \simeq 0.3$; $R_c \simeq 0.68 \times 10^6$

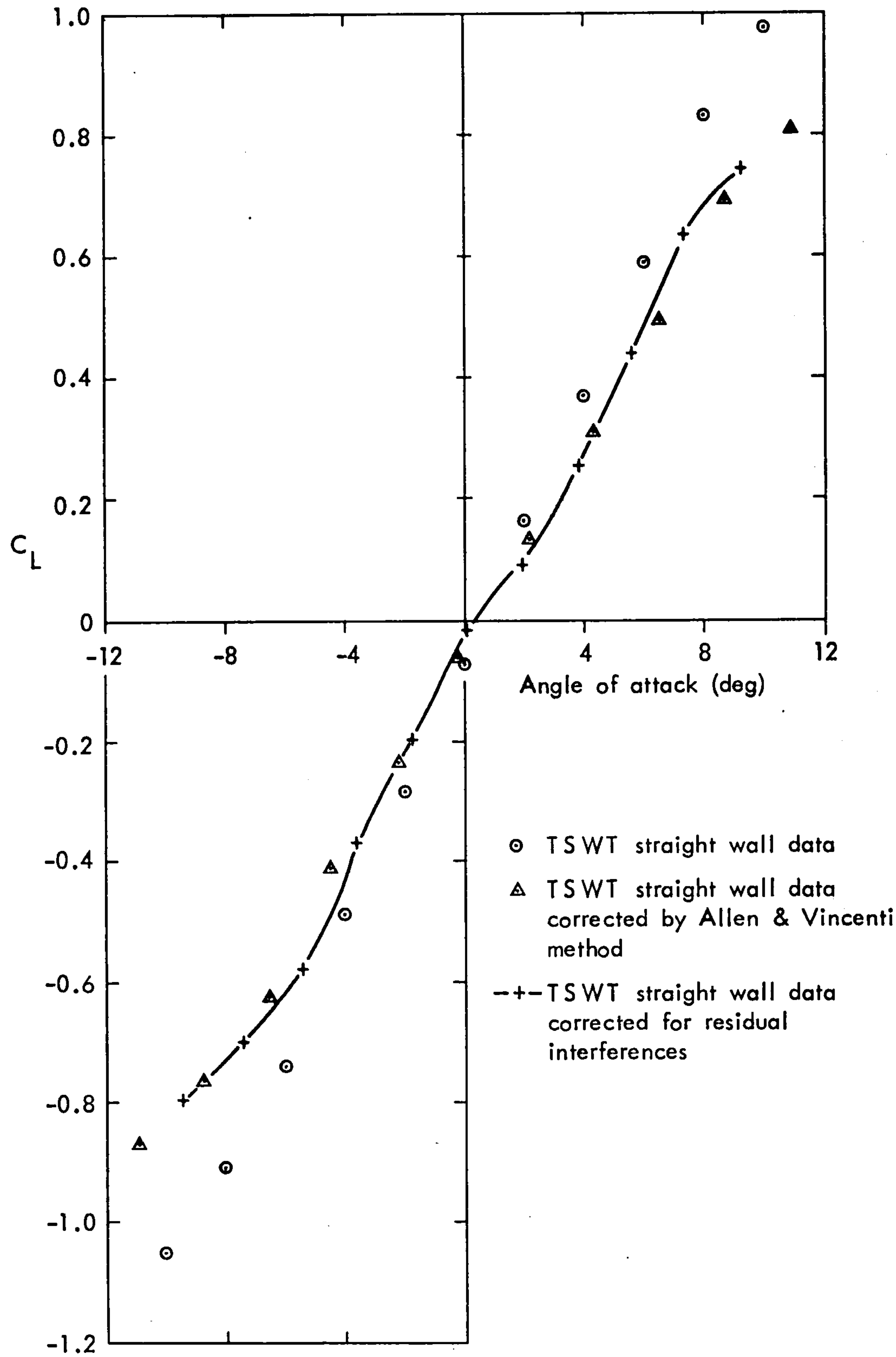


FIG. 4(c) LIFT CURVE SLOPES ; $M_\infty \simeq 0.3$

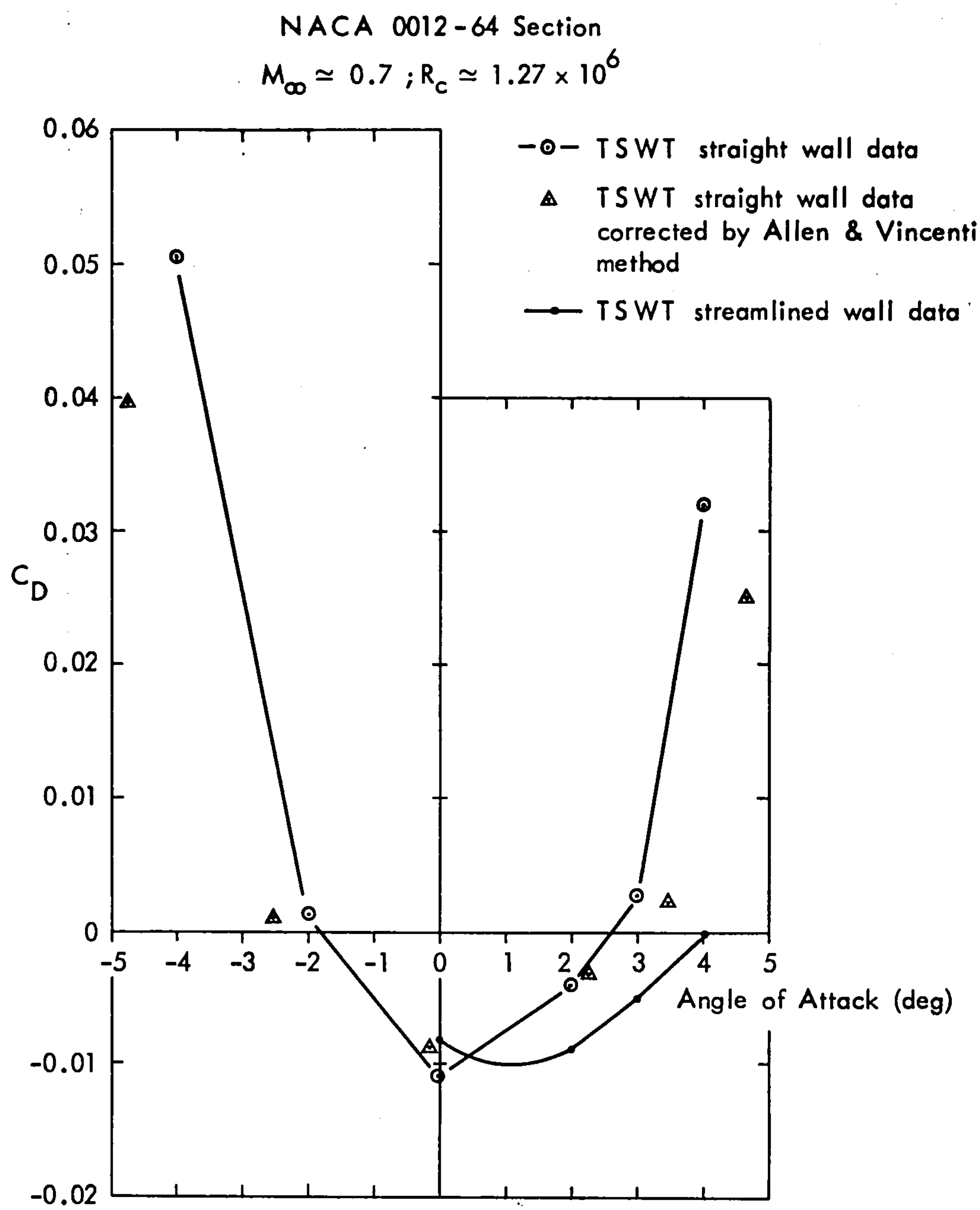


FIG. 5(a) VARIATION OF MODEL PRESSURE DRAG WITH ANGLE OF ATTACK ; $M_\infty \simeq 0.7$

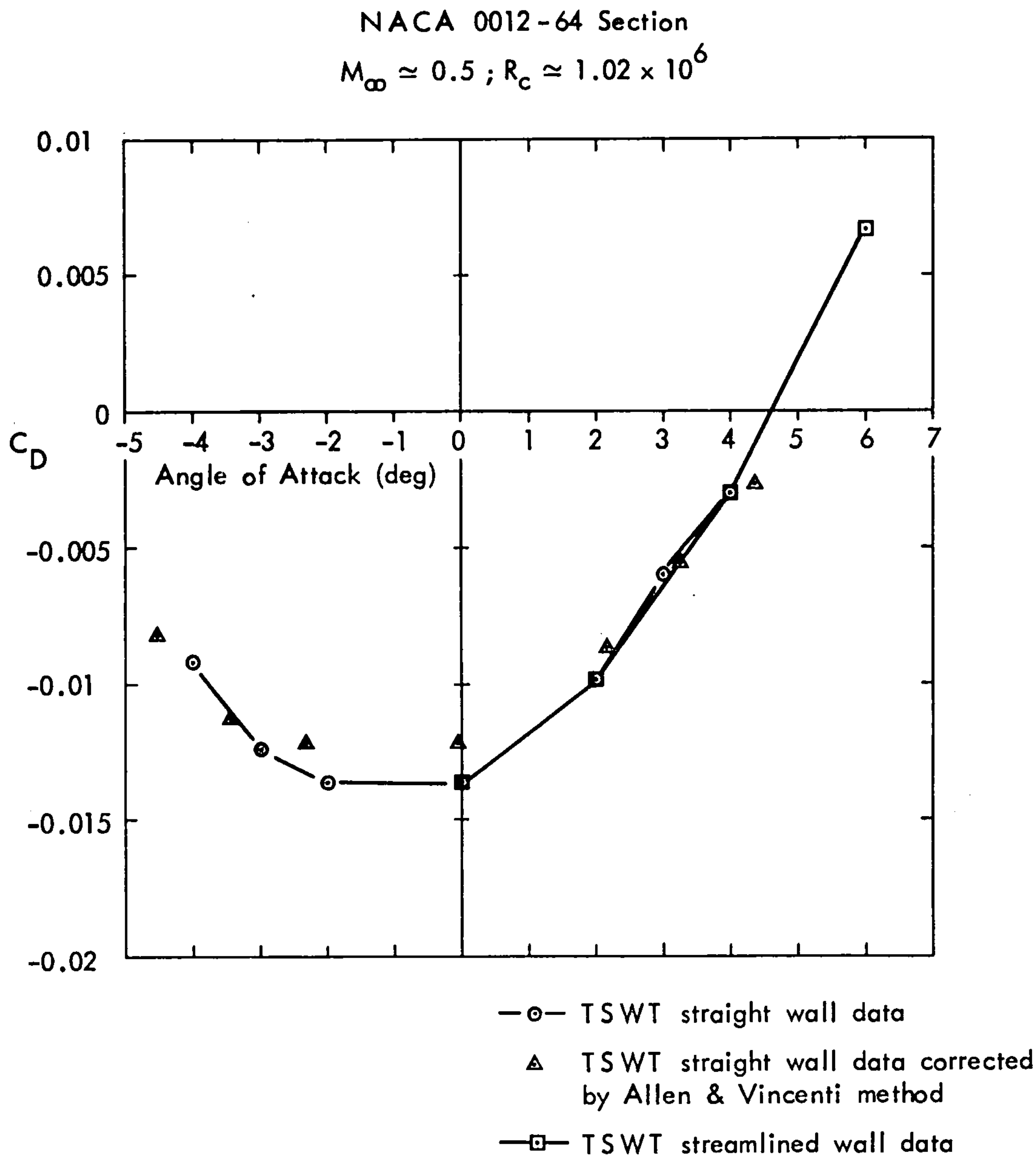


FIG. 5(b) VARIATION OF MODEL PRESSURE DRAG WITH ANGLE OF ATTACK ; $M_\infty \simeq 0.5$

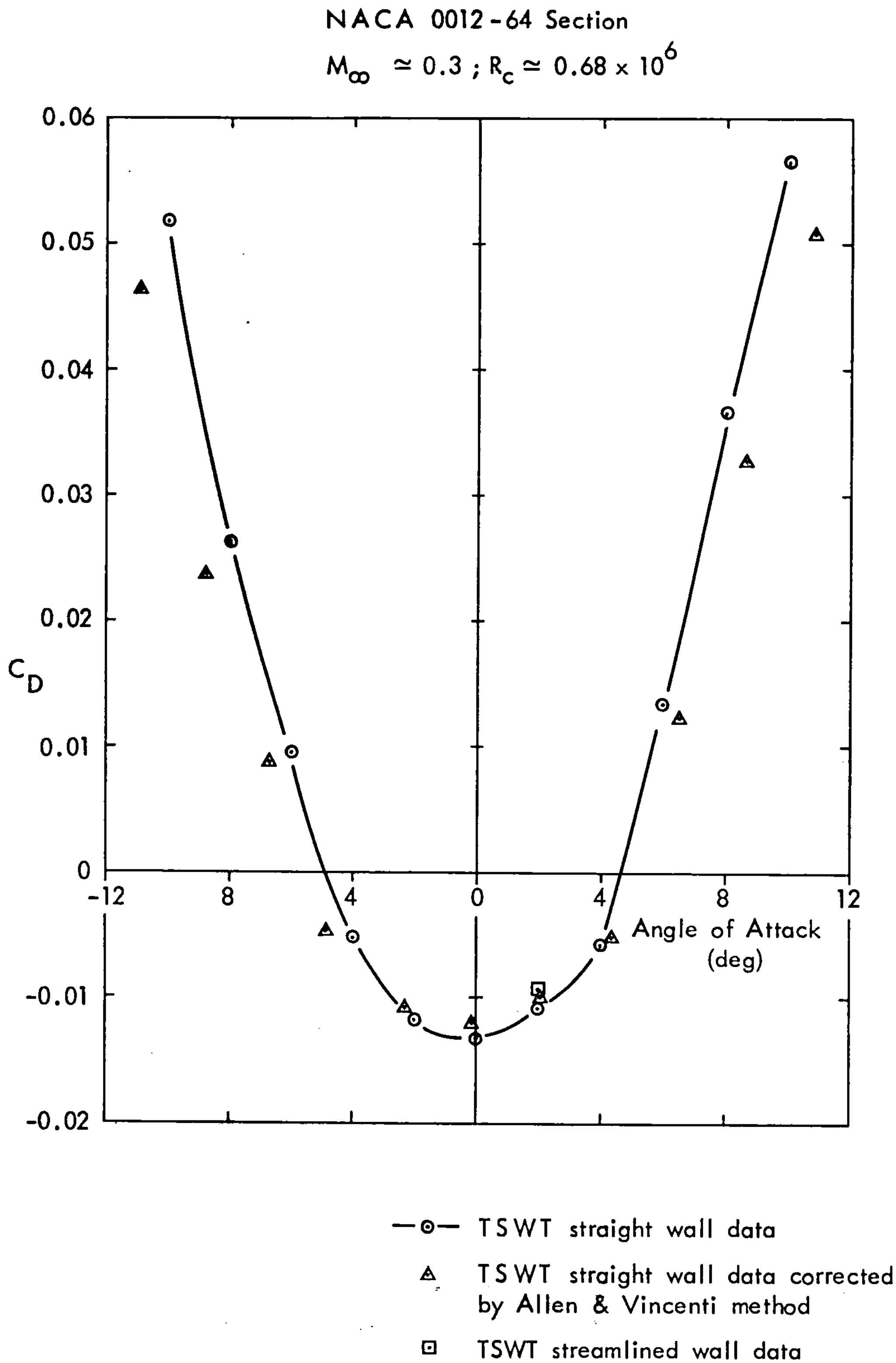


FIG. 5(c) VARIATION OF MODEL PRESSURE DRAG WITH ANGLE OF ATTACK ; $M_\infty \simeq 0.3$

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15. Supplementary Notes

Langley technical monitor: Charles L. Ladson
Semi-annual Progress Report for the period to October 1980. The Principal Investigator was Dr. M. J. Goodyer.

16. Abstract

This is a semi-annual progress report for the period from May 1980 to October 1980 on work undertaken on NASA Grant NSG-7172 entitled "The Self-Streamlining of the Test Section of a Transonic Wind Tunnel." The Principal Investigator is Dr. M. J. Goodyer. Included in this report are data for an NACA 0012-64 airfoil at Mach number of 0.3, 0.5, and 0.7 with the test section flexible walls set in the "straight" position. These data are presented because they may prove useful in the development of wind tunnel correction techniques.

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