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AERODYNAMIC DATA FROM A TWO-DIMENSIONAL CAMBERED AIRFOIL SECTION IN A SHALLOW TRANSONIC FLEXIBLE WALLED TEST SECTION

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1. Introduction

Further work to validate the flexible wall technique in two-dimensional testing has been carried out with the Transonic Self-Streamlining Wind Tunnel[1] (TSWT) using a cambered NPL 9510 section, larger and perhaps of more challenging design than the NACA 0012-64 section previously tested[1].

Model data on lift and drag was obtained over a Mach number range up to 0.87 and at angles of attack from zero to 6°. The results taken with the walls streamlined were then compared with two sources of reference data obtained in conventional slotted walled transonic test sections. The reference data cannot be considered interference free but is the best currently available at low Reynold's numbers, and has to provide a basis for assessing the quality of TSWT data.

There were 52 runs of the test section in carrying out this programme, and some of the streamlining cycles[1] were performed using an automated wall control system[1,2] linked to a mini-computer. These runs provided further useful TSWT operational experience with a larger model than previously tested. Limits to both test Mach number and model angle of attack were found.

NPL 9510 data from TSWT is presented as a library of numerical and graphical information which may prove useful to others engaged in the evaluation, design and use of transonic flexible walled test sections.

2. Model Description

The two dimensional model used in the TSWT tests described here was an NPL 9510 section of 15.24cm (6 inches) chord and 20.32cm (8 inches) span, constructed of HP 9-4-20 ALLOY Steel. The section surface co-ordinates are shown on Table 1 and the section profile is plotted on Figure 1.

Surface pressure tappings were positioned over the mid-span portion of the model on both surfaces. The tappings are concentrated over specific parts of the airfoil profile, namely the 50% chord region on the upper surface and the trailing edge region on the lower surface as shown on Table 2. Tests were performed with and without a transition strip positioned around the leading edge to about 3% chord.

Since the test section width was 15.24cm (6 inches), the tips of the model were buried in purpose-built window blanks which acted as the model supports (See Figure 2). No schlieren photography was possible with this arrangement.

3. Reference Data

In order to reduce the number of uncertainties when evaluating the data from TSWT tests, lift and drag data was obtained on the same model in the NASA Langley Research Center (LRC) 0.3-Meter Transonic Cryogenic Tunnel. Its slotted two-dimensional test section gave a height/chord ratio just greater than two. The LRC tests* were performed at a stagnation pressure above ambient and at a stagnation temperature somewhat below ambient and in nitrogen, which together resulted in chord Reynolds numbers being about 66% higher than in TSWT at the same Mach number. Transition strips were installed from 4- to 6 percent chord on the upper surface and from 6- to 8 percent chord on the lower surface for the LRC tests.

Reference lift and drag data was also available from original NPL tests[2] for comparison. This data was obtained from a 25.4cm (10 inch) chord model in an NPL transonic tunnel fitted with a two-dimensional slotted test section with a height equal to 3 chords. The test were performed at ambient stagnation conditions giving chord Reynolds numbers also about 66% greater than for TSWT. A transition band was fitted to the lower surface of the model from 6-8% chord for all NPL tests and also for the majority of those tests from 4-6% chord on the upper surface.

When comparing TSWT data with that from LRC and NPL, it should be noted that:

1. The reference data is not corrected for any boundary interference and therefore where possible the pressure distributions are compared with the model C_Ns nearly matched, to remove uncertainties about angle of attack.
2. There are differences in the chord Reynolds numbers.

In view of this situation, the reference data is only used here as an indication of model performance.

For both sets of reference data, lift was obtained from integrated pressure distributions. Drag was obtained from conventional wake traverses made 0.736 chord downstream of the trailing edge in the LRC tests and one chord downstream in the NPL tests. All reference drag data presented here was obtained from traverses down the tunnel centerline.

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*Unpublished Work

4. Transonic Self-Streamlining Wind Tunnel Data

4.1. Lift

A total of fifty-two sets of data were acquired with the walls 'Streamlined'. Twenty-one points were with no transition strip fitted to the model (as listed on Table 3) and thirty-one were with the transition strip fitted (as listed on Table 4) to observe its effect on model performance. The model pressure distribution for each test point is tabulated in Table 5 and plotted on Figure 3 in the order shown on Tables 3 and 4.

The TSWT lift data is summarised in the plots of the normal force coefficient C_N versus angle of attack for approximately freestream Mach numbers of 0.5, 0.6, 0.7, 0.75 and 0.8 shown on Figure 4. Both transition fixed and transition free data is shown. The reference data is also shown for comparison. The NPL data is plotted conveniently as lift coefficient C_L which is little different from C_N at the moderate angles of attack discussed here.

For $M_\infty \simeq 0.5$, there is a small difference between the normal force curve slope ($dC_N/d\alpha$) for the TSWT data (transition fixed) and LRC data. However the TSWT data (transition free) shows better agreement at lower angles of attack. The ratios of the two TSWT curve slopes with the LRC slope have the values:

Transition Fixed : 1.10
Transition Unfixed: 1.04

over the angle of attack range $0^\circ \leqslant \alpha \leqslant +6^\circ$.

Reynolds number effects could account for some of this difference.

The NPL data shows a consistent shift in angle of attack relative to the other data sets but the slope compares favourably with the TSWT result.

For $M_\infty \simeq .6$, there is agreement between LRC and TSWT data (transition fixed). The NPL data is again displaced by an amount corresponding to an angle of attack of roughly half a degree.

For $M_\infty \simeq .7$, there is good agreement between LRC and TSWT data. The NPL data at low angles is again displaced, but has a slope roughly equal to that of the TSWT data. At the higher angles the slope is seen to increase and the data diverge from the other two sources.

For $M_\infty \simeq .75$, the TSWT data (transition fixed) shows the same trend as the reference data, an increasing lift curve slope with angle of attack. A disparity between TSWT and LRC data appears at the higher angles of attack, while the NPL data diverges more strongly.

For $M_\infty \simeq .8$, the values of C_N from the TSWT tests (transition fixed) compares favourably with LRC data above about 1° angle of attack, see figure 4e. There is however, a discrepancy between LRC and TSWT data at $\alpha = 0^\circ$. At this Mach number, the shock positions are sensitive to the boundary layer condition and there is a correspondingly large difference in model performance for the TSWT tests with transition fixed and free, as clearly shown in the model pressure distributions for the test case $M_\infty \simeq .8$; $\alpha = 3^\circ$ shown on Figure 5. The upper surface shock is shown to travel from about 60% chord, transition free to about 45% chord, transition fixed. It is apparent that the TSWT data (transition free) is substantially different from the other data sources at high subsonic Mach numbers. The NPL data at $M_\infty \simeq .8$ shows a disparity with both TSWT and LRC equivalent to up to half a degree in angle of attack. There is also a pronounced reduction of the lift curve slope in the NPL data beyond about $\alpha = 2^\circ$, not evident in the TSWT data. This perhaps indicated an earlier stall due to a larger effective angle of attack of the NPL model.

TSWT data was obtained at higher Mach numbers over only a limited angle of attack range, partly to limit loads. The intention was to locate the important limit to test Mach number giving a breakdown in the wall setting strategy in the manner previously reported[1]. The highest Mach number at which wall streamlining was acheived was 0.87 with $\alpha = 2^\circ$. It is interesting to observe the variation of C_N over the Mach number band 0.5 to 0.87 at this angle of attack shown on Figure 6. A shock stall is evident at about Mach 0.85. Again there is reasonable agreement with LRC data as far as it goes. Evidence of an angle of attack error is visible in the NPL data which is above the remainder up to the shock stall. The shift of the onset of shock stall from Mach 0.85 in TSWT to Mach 0.79 in the NPL tests is also indicative of a disparity in angle of attack.

Further detailed comparisons of TSWT results with reference data have been made. The model pressure distributions for the test case $M_\infty \simeq .7$; $\alpha = 4^\circ$ are shown on Figure 7 from TSWT data (transition fixed) and the LRC data sets. The model C_Ns are not perfectly matched but the upper surface shock is roughly in the same position for both tests. However, the pressure recovery downstream of this shock is different for the two tests perhaps due to different thicknesses of the model's boundary layer during each test. The disparities in the pressure distributions on the lower surface could be similarly caused. The peak Mach number on the top wall was 0.82.

For the test case $M_\infty \simeq .75$; $\alpha = 2^{\circ}$ the upper surface shock may be slightly misplaced forwards by some 5% chord in the TSWT tests, as shown on Figure 8. The suction peak obtained in the LRC tests is slightly lower than the TSWT result, which may have been caused by different grit concentrations on the leading edge. A comparison between TSWT and NPL data for the same test case, shown on Figure 9, illustrates similar orders of differences. The upper surface shock position is matched but there are discrepancies downstream of the shock and on the lower surface. Unfortunately, using the available data, lift coefficients could not be better matched between the TSWT and NPL tests.

The peak top wall Mach number in this TSWT test was 0.837. At higher freestream Mach numbers the model shocks extended towards the walls such that for the cases: $M_\infty = .8$; $\alpha = 2^{\circ}$ and $M_\infty = .87$; $\alpha = 2^{\circ}$ the peak Mach numbers on the top wall were 0.964 and 1.087 respectively. The need for some shaping of the wall to absorb the thickening of its boundary layer under the shock boundary layer interaction has been demonstrated in previous tests[1,3], but further work is required before this procedure can be followed on a regular basis.

The repeatability of results has been investigated. For the case $M_\infty \simeq .7$; $\alpha = 2^{\circ}$ (transition free) two values of C_L were obtained from different streamlining paths. One streamlining cycle was initiated with the walls set to the $M_\infty = .7$; $\alpha = 1^{\circ}$ streamlined contours, requiring only one iteration for streamlining and giving $C_L = .4589$. The other cycle was initiated with the walls set to the $M_\infty = .7$; $\alpha = 0^{\circ}$ streamlined contours. Here three iterations were required giving $C_L = .4478$. There is a difference of .0111 (2.5%) in C_L. The residual errors were greater than previously reported values[1] (α error $< 0.015^{\circ}$) which may have worstened this comparison. For the test case of $M_\infty = .7$; $\alpha = 0^{\circ}$ (transition fixed) a repeat run was performed (Run 390) with the walls reset to the Run 380 streamline contours after some routine streamlining cycles over a range of angle of attack. The model C_L between the two tests reduced by 0.0123 (8%) with a corresponding small change in the residual wall-induced α error from -0.011 to -.0079. The absolute values of the differences in coefficients is indicative of the repeatability of the tunnel system. This includes the effect of repeatability in setting angle of attack which is not claimed to be better than $\pm 0.1^{\circ}$. A setting error of this magnitude would itself introduce an error in C_N of about 0.013 with this model, the same order as the figures observed.

4.2. Drag

Using a previously reported Wake Traverse technique[4], drag data was obtained on the NPL 9510 model for a limited number of tests points, over the Mach number range 0.5 to 0.8.

The traversing plane was 1.083 chords downstream of the model trailing edge, on the tunnel centerline. A total of thirteen traverses were performed. The resulting drag coefficient data is plotted on Figure 10, compared with the LRC and NPL reference data, for approximate freestream Mach numbers of 0.5, 0.6, 0.7, 0.75 and 0.8.

At $M_\infty \simeq .5$, the TSWT data is in good agreement only at $\alpha = 0^\circ$. From the C_N data shown on Figure 4a, it is possible to identify an angle of attack error for the TSWT data at $\alpha = 4^\circ$. This would have the effect of shifting the data sets closer together. At $\alpha = 2^\circ$ there is a significant difference, part of which may be the effect of ill-defined edges of the wake observed in this test. Unfortunately, the freestream Mach number at the traversing plane is only known to be approximately that of the reference freestream. The NPL data is misplaced from the LRC results by a roughly equal amount in common with the lift drag.

At $M_\infty \simeq .6$ there is reasonable agreement particularly with the NPL data and likewise at $M_\infty \simeq .7$ and $M_\infty \simeq .75$ albeit over a reduced angle of attack range. The LRC and NPL data at $M_\infty \simeq .8$ is scattered and TSWT drag data is shown to lie below the reference data.

The best that can be claimed is that these results are plausible - in common with previously reported data[4]. However, there remains the problem of defining the edges of the wake. The choice of this value has been found to have a significant effect on the derived value of C_D. The lack of agreement between the sources of data shown on Figure 10 may just illustrate the discrepancies found between results from different wind tunnels.

5. Streamlining Performance

Fifty-one streamlining cycles have been performed in TSWT with the NPL 9510 airfoil. Experience was gained on the effects of modifications of the wall setting strategy on the number of iterations required in streamlining. The modifications were to the overshoot factors (a_3 and a_4 in reference 5).

Only three streamlining cycles were initiated with straight walls for the test conditions of $M_\infty = .5$; $\alpha = 0^\circ$ and $M_\infty = .7$; $\alpha = 0^\circ$, each requiring an average of four iterations with the overshoot factors equal to 0.8. All other streamlining cycles were initiated with the walls contoured to known shapes usually streamline contours from a previous cycle. No straight wall tests at significantly higher angles of attack or higher speeds were possible due to test section choking caused by the high blockage of the model. For example, the test section choked at the condition $M_\infty = .7$; $\alpha = 0^\circ$. The boundary interference which is present during a straight wall run is shown in the model pressure distributions on Figure 11 with the walls set straight and streamlined for the test case $M_\infty = .5$; $\alpha = 0^\circ$.

For the sets of data acquired at Mach 0.5, 0.7, 0.75 and 0.8 the Mach number was held constant while the model angle of attack was increased in convenient 1° steps from 0° to a maximum angle of attack determined by limits to wall movement. Test Mach number was then increased with the model set to zero angle of attack while the walls remained fixed.

At Mach 0.5 it was found that an average of one iteration was required per streamlining cycle with the overshoot factors set at 0.8, while an average of two iterations was required with the overshoot factors equal to 0.6. At Mach 0.7, the average number of iterations increased to two with the overshoot factors set to 0.8. Changing the factors to 0.6 produced an average of three iterations. At Mach 0.75 the average number of iterations reduced to two with the overshoot factors set to 0.6. At Mach 0.8, there was an average of three iterations with the overshoot factor set at 0.5 or 0.6. However, the tests with a_3 and a_4 equal to 0.5 may not have determined good wall streamlines since one measure of streamlining quality, E, was not reduced to an acceptable level in these tests, although the residual interferences were acceptably small.

A series of tests were performed at $M_\infty = .6$, with the angle of attack decreased from 5° to 0° in 1° steps. The average number of iterations per streamlining cycle for the series was two with a minimum of one. These results were acheived with the overshoot factors equal to 0.6.

While no firm conclusions can be made from an exploration having such a narrow scope, enough was done in this series and previously to indicate the following trends in the variations in number of iterations to streamline: an overshoot factor of 0.8 is generally superior to lower values, iterations tend to increase with Mach number. The number of iterations appears dependent on the change in model lift between test conditions. Altering the overshoot factors may have the effect of tailoring the test section to a particular model. It may be better to plan a test programme based on expected model performance and limit ΔC_L between test conditions. This limit on C_L will presumably be some function of test section height and model size.

For the most critical test case, $M_\infty = 0.87$; $\alpha = 2^o$, the wall setting strategy began to break down, as indicated by numerous iterations required in the hunt for streamlines. The set of contours which were finally reached coincided with E reaching a minimum. However, the residual interferences had still not reached the usual low levels[1]. At this particular test condition, there was a supercritical patch of flow reaching the top wall having a peak wall Mach number of 1.087. This test condition compares, in terms of this Mach number with the case encountered during previously reported tests[1] with a smaller model where strategy breakdown was observed at $M_\infty = 0.89$, $\alpha = 4^o$.

Streamlining was a routine operation for all Mach numbers up to 0.8. To achieve higher speeds, it was necessary to introduce changes in the tunnel operating procedure to generate the required test Mach number in the manner already reported[1].

A family of wall contours covering a range of angle of attack for an approximate freestream Mach number of 0.7, are shown on Figure 12. These are streamlined with contours, showing the strong effects of a large model and its attitude on test section shape. There is a reasonably linear increase in the movement of the walls apart with increasing angle of attack.

The change of upwash with lift is apparent ahead of the model, with the opposite effect downstream. It should perhaps be re-emphasised that the walls take up these shapes quite automatically, in response to measurements made only at the walls.

The variation of streamline wall contours with Mach number is shown on Figure 13 over the Mach number range 0.5 to 0.87 for $\alpha = 2^o$. An effect of compressibility is visible in the walls moving apart in the region of the model, progressively more rapidly as Mach number is increased above 0.7. It is interesting to note that at Mach 0.87 the walls in the region of the model have moved apart by an amount roughly equal to the model thickness.

For the test case $M_\infty \simeq .8$; $\alpha = 3^\circ$ there was a significant difference between TSWT C_L data with transition fixed and unfixed (see Figure 5) although the walls were streamlined in both cases. The two sets of contours are shown on Figure 14, they are significantly different and are supporting evidence that the flow round the model was different in the two cases. The upwash ahead of the model is shown by the walls to have changed in the same sense as the model lift between the two runs.

For tests at Mach 0.8 and below, the only limitation on model angle of attack was the available wall movement. This limit is reached with the existing hardware at the following test conditions: $M_\infty = 0.5$, $\alpha = 6^\circ$; $M_\infty = 0.7$, $\alpha = 5^\circ$; $M_\infty = 0.75$, $\alpha = 4^\circ$. The severity of the wall movement required for streamlining is clearly shown on Figure 2 for the case $M_\infty = 0.87$, $\alpha = 2^\circ$. More wall movement than the current restrictions allow (limited by transducer stroke at present) is available should it be required.

The NPL 9510 tests have provided further useful experience with the on-line control system[1,2]. Streamlining cycles were performed rapidly under computer control with wall setting times of order minutes, one iteration typically taking thirty seconds. In fact, thirty streamlining cycles were completed in less than the time it took to perform the first ever streamlining cycle at Southampton in 1973 - two working weeks!

6. Principal Conclusions

1) Wall streamlining has been routinely performed around a cambered two-dimensional airfoil with a test section height to model chord ratio of unity, up to a freestream Mach number of 0.8.

2) The wall setting strategy has been observed to breakdown at the test point $M_\infty = .87$; $\alpha = 2^\circ$ during these particular tests.

3) Model lift from the TSWT tests compares favourably with the LRC reference data over the Mach number range 0.5 to 0.8 despite the use of a shallow non-porous test section and a disparity in Reynolds number.

4) Model drag derived from TSWT tests is reasonable, but claims for its absolute accuracy cannot yet be made.

5) The number of iterations per streamlining cycle can be reduced to one by fine tuning of the wall setting strategy when used with one particular model. However, this approach may not be applicable to a production facility.

List of Symbols

α	-	Model angle of attack
CC	-	Chordwise force coefficient
CD	-	Pressure drag coefficient
C_D	-	Drag coefficient
C_L	-	Lift coefficient
CM	-	Pitching moment coefficient about the leading edge
C_N	-	Normal force coefficient
C_p	-	Pressure coefficient
C_p^*	-	Sonic pressure coefficient
M_∞	-	Freestream Mach number
R_c	-	Chord Reynolds number
X	-	Chordwise position from the leading edge
Y_s	-	Model surface displacement up from the leading edge

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Table 1

Co-ordinates of the NPL-9510 Section

X_c	Y_{upper}	Y_{lower}
.0	-.0001	-.0001
.0024	.0235	-.0238
.0096	.0456	-.0477
.0150	.0565	-.0586
.0300	.0766	-.0812
.0600	.1002	-.1108
.0900	.1153	-.1321
.1200	.1260	-.1499
.1500	.1343	-.1652
.1800	.1414	-.1790
.2400	.1532	-.2033
.3000	.1633	-.2243
.3600	.1723	-.2426
.4200	.1804	-.2586
.4800	.1877	-.2728
.5400	.1945	-.2852
.6000	.2006	-.2964
.7200	.2118	-.3159
.8400	.2216	
.9600	.2303	-.3455
1.0800	.2382	
1.2000	.2454	-.3668
1.3200	.2523	
1.4400	.2589	-.3804
1.5600	.2650	
1.6800	.2704	-.3863
1.8000	.2755	
1.9200	.2801	-.3845
2.0400	.2845	
2.1600	.2884	-.3748
2.2800	.2920	
2.4000	.2950	-.3577
2.5200	.2975	
2.6400	.2994	-.3332
2.7600	.3008	
2.8800	.3016	-.3021
3.0000	.3019	
3.1200	.3017	-.2657
3.2400	.3010	
3.3600	.2997	-.2257
3.4800	.2982	
3.6000	.2958	-.1842
3.7200	.2927	
3.8400	.2891	-.1424
3.9600	.2845	
4.0800	.2793	-.1023
4.2000	.2726	
4.3200	.2649	-.0638
4.4400	.2558	
4.5600	.2456	-.0302
4.6800	.2341	
4.8000	.2214	-.0028
4.9200	.2075	
5.0400	.1923	+.0175
5.1600	.1760	
5.2800	.1585	+.0285
5.4000	.1402	
5.5200	.1205	+.0291
5.6400	.0993	
5.7600	.0767	+.0193
5.8800	.0533	
6.0000	.0294	+.0001

(All Co-ordinates in Inches)

Table 2

Measured Co-ordinates for Pressure Ports

NPL 9510 Section

X_c	Y_{upper}	X_c	Y_{lower}
0.0	0.0	0.0	0.0
.0451	.0925	.0598	-.1100
.0605	.1027	.1061	-.1417
.0986	.1207	.3158	-.2292
.1202	.1282	.6137	-.2986
.1591	.1386	.9160	-.3407
.1845	.1444	1.2122	-.3675
.2186	.1513	1.6622	-.3861
.3108	.1671	2.2539	-.3690
.4613	.1876	2.7041	-.3255
.6111	.2036	3.1540	-.2602
.9106	.2288	3.6039	-.1839
1.2116	.2482	4.0529	-.1062
1.5111	.2646	4.3038	-.0662
1.8109	.2780	5.1044	+.0215
2.4109	.2975	5.4057	+.0302
2.7116	.3024	5.7046	+.0227
3.0117	.3039	6.0000	+.0147
3.1617	.3035		
3.3121	.3024		
3.4621	.3005		
3.6122	.2976		
3.7617	.2937		
3.9116	.2886		
4.0621	.2822		
4.2118	.2740		
4.5098	.2522		
4.8102	.2223		
5.1116	.1847		
5.4070	.1412		
5.7108	.0884		
6.0000	.0147		

(All Co-ordinates in Inches)

Table 3

Summary of TSWT Data

NPL 9510 (Transition free)

Figure No.	Run No.	Model α (deg)	Mach No.	Iterations from straight walls	Iterations from contoured walls	Change from start contours		A3	E_{AV}	C_L
						Δα	ΔM			
1	340	0.0	.853	-	7	0	0.05	.5	.0108	.0899
2	350	3.0	.805	-	4	1.0	0	.5	.016	.8167
3	345	2.0	.804	-	4	2.0	0	.5	.012	.6273
4	353	1.0	.798	-	2	-1.0	0	.5	.007	.3172
5	332	0.0	.809	-	2	0	0.07	.5	.0052	.1084
6	329	0.0	.739	-	2	-4.0	0.14	.7	.0065	.1437
7	313	4.0	.699	-	2	1.0	0	.5	.0084	.7421
8	302	3.0	.7	-	3	1.0	0	.5	.0088	.6061
9	296	2.0	.702	-	1	1.0	0	.8	.0068	.4589
10	280	2.0	.702	-	3	2.0	0	.8	.0042	.4478
11	294	1.0	.699	-	3	1.0	0	.8	.0048	.2957
12	275	0.0	.702	4	-	-	-	.8	.0072	.1702
13	316	4.0	.599	-	2	0	-0.1	.5	.0097	.6524
14	318	4.0	.588	-	1	0	-.05	.8	.0092	.6456
15	324	6.0	.5	-	2	1.0	0	.7	.0032	.7621
16	321	5.0	.505	-	1	1.0	0	.7	.0048	.6743
17	306	4.0	.502	-	1	1.0	0	.8	.0068	.6047
18	304	3.0	.502	-	1	0	-0.2	.8	.0036	.5144
19	301	2.0	.503	-	1	2.0	0	.8	.0068	.4086
20	356	1.0	.497	-	2	1.0	0	.7	.0039	.2744
21	288	0.0	.506	3	-	-	-	.8	.0073	.1679

Table 4

Summary of TSWT Data

NPL 9510 (Transition fixed)

Figure No.	Run No.	Model α (deg)	Mach No.	Iterations from straight walls	Iterations from contoured walls	Change from start contours		A3	E_{AV}	C_L	Residual α error (deg)
						Δα	ΔM				
22	398	2.0	.87	-	0	0	0.02	.7	.0056	.4635	-.051
23	396	2.0	.849	-	-	1.0	0.005	.7	.0054	.512	-.0055
24	395	1.0	.844	-	-	1.0	0	.7	.0052	.2581	.0075
25	394	0.0	.837	-	-	-5.0	0.237	.7	.0073	.0274	-.0059
26	389	3.0	.804	-	4	1.0	0	.6	.0068	.7073	-.0045
27	388	2.0	.802	-	4	1.0	0	.6	.0061	.5011	-.0088
28	387	1.0	.802	-	2	1.0	0	.6	.0035	.2654	.0095
29	386	0.0	.801	-	-	-	-	.6	.0074	.099	.0058
30	393	0.0	.753	-	1	0	0.01	.7	.0048	.15	-.0121
31	391	3.0	.758	-	3	1.0	0	.7	.0053	.6694	.01
32	403	2.0	.749	-	2	2.0	0	.6	.004	.4675	.0006
33	402	0.0	.743	-	2	0	0.043	.7	.0044	.1603	-.0021
34	384	4.0	.696	-	3	1.0	0	.6	.0027	.7026	-.0068
35	383	3.0	.696	-	3	1.0	0	.6	.0042	.5884	.0103
36	382	2.0	.701	-	2	1.0	0	.6	.006	.4292	.0174
37	381	1.0	.697	-	4	1.0	0	.6	.0025	.2873	-.0084
38	380	0.0	.704	-	1	0	0.1	.6	.006	.1499	-.011
39	390	0.0	.704	-	-	0	0	.6	.0028	.1376	-.0079
40	374	5.0	.602	-	3	0	0.1	.6	.0065	.7097	.0018
41	375	4.0	.598	-	1	-1.0	0	.6	.005	.6203	.0061
42	376	3.0	.605	-	3	-1.0	0	.6	.0035	.5359	-.0127
43	377	2.0	.598	-	1	-1.0	0	.6	.0069	.3942	.0096
44	378	1.0	.6	-	2	-1.0	0	.6	.0039	.2799	.0075
45	379	0.0	.595	-	1	-1.0	0	.6	.0054	.155	.0069
46	371	5.0	.508	-	2	1.0	0	.6	.0053	.6927	.0045
47	373	4.9	.501	-	2	-1.0	0	.6	.004	.6828	.0069
48	370	4.0	.501	-	2	1.0	0	.6	.005	.6339	.0068
49	369	3.0	.493	-	2	1.0	0	.6	.0045	.493	.0044
50	368	2.0	.5	-	1	1.0	0	.6	.0056	.406	.012
51	367	1.0	.498	-	1	1.0	0	.6	.0056	.2658	-.0086
52	366	0.0	.496	4	-	-	-	.5	.051	.1431	.0008

Table 5 NPL 9510 Pressure Distributions

NPL SECTION ANALYSIS
9510

RUN NO. = 340

ALPHA = 0.0

MACH NO. =0.8531

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

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.1288	0.0	1.1429
0.7	-0.1798	1.0	0.5501
1.0	-0.4413	1.8	0.2819
1.6	-0.6400	5.3	-0.1362
2.0	-0.7412	10.2	-0.4360
2.6	-0.7494	15.3	-0.5414
3.1	-0.7973	20.2	-0.6168
3.6	-0.7740	27.7	-0.7278
5.2	-0.7125	37.6	-0.7155
7.7	-0.6496	45.1	-0.5935
10.2	-0.5853	52.6	-0.4089
15.2	-0.5388	60.1	-0.2854
20.2	-0.4704	67.5	-0.1741
25.2	-0.1866	71.7	-0.1092
30.2	-0.2868	85.1	0.0632
40.2	-0.3746	90.1	0.1249
45.2	-0.4666	95.1	0.1488
50.2	-0.4775	100.0	0.1071
52.7	-0.4719		
55.2	-0.4854		
57.7	-0.4732		
60.2	-0.5057		
62.7	-0.5043		
65.2	-0.5333		
67.7	-0.5795		
70.2	-0.6106		
75.2	-0.5218		
80.2	-0.2659		
85.2	-0.1776		
90.1	-0.0827		
95.2	0.0186		
100.0	0.0963		

	UPPER	LOWER	TOTAL
CN	0.3858	-0.2959	0.0899
CC	-0.0021	0.0287	0.0266
CM	-0.1618	0.0972	-0.0646

AIRFOIL PERFORMANCE

CL	CD	CM
0.0899	0.0266	-0.0646

Table 5.1.

NPL SECTION ANALYSIS
9510

RUN NO. = 350

ALPHA = 3.0

MACH NO. =0.8046

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 22

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0352	0.0	1.0378
0.7	-0.7211	1.0	0.8323
1.0	-0.9777	1.8	0.5943
1.6	-1.1374	5.3	0.1966
2.0	-1.2262	10.2	-0.0547
2.6	-1.2146	15.3	-0.1517
3.1	-1.2990	20.2	-0.2298
3.6	-1.2801	27.7	-0.3166
5.2	-1.2482	37.6	-0.3093
7.7	-1.2102	45.1	-0.2356
10.2	-1.1169	52.6	0.0192
15.2	-1.1475	60.1	0.1271
20.2	-1.1257	67.5	0.2407
25.2	-1.1255	71.7	0.2967
30.2	-1.1299	85.1	0.3849
40.2	-1.0889	90.1	0.3833
45.2	-1.1357	95.1	0.3211
50.2	-1.0670	100.0	0.0795
52.7	-1.0263		
55.2	-1.0119		
57.7	-0.9367		
60.2	-0.7936		
62.7	-0.5406		
65.2	-0.4182		
67.7	-0.3067		
70.2	-0.2700		
75.2	-0.2925		
80.2	-0.2877		
85.2	-0.2327		
90.1	-0.1504		
95.2	-0.0360		
100.0	0.0784		

	UPPER	LOWER	TOTAL
CN	0.7570	0.0603	0.8173
CC	-0.0286	0.0193	-0.0093
CM	-0.2681	-0.0701	-0.3382

AIRFOIL PERFORMANCE

CL	CD	CM
0.8167	0.0335	-0.3382

Table 5.2.

NPL SECTION ANALYSIS
9510

RUN NO. = 345

ALPHA = 2.0

MACH NO. =0.8042

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 17

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0699	0.0	1.0804
0.7	-0.5927	1.0	0.7370
1.0	-0.8669	1.8	0.4854
1.6	-1.0393	5.3	0.0696
2.0	-1.1313	10.2	-0.1880
2.6	-1.1225	15.3	-0.2780
3.1	-1.2043	20.2	-0.3588
3.6	-1.1794	27.7	-0.4618
5.2	-1.1401	37.6	-0.4324
7.7	-1.0934	45.1	-0.3662
10.2	-1.0016	52.6	-0.0458
15.2	-1.0234	60.1	0.0872
20.2	-1.0016	67.5	0.2142
25.2	-0.9965	71.7	0.2743
30.2	-0.9965	85.1	0.3724
40.2	-0.8809	90.1	0.3725
45.2	-0.9204	95.1	0.3175
50.2	-0.8283	100.0	0.0975
52.7	-0.6085		
55.2	-0.3651		
57.7	-0.2651		
60.2	-0.3086		
62.7	-0.3463		
65.2	-0.3987		
67.7	-0.4653		
70.2	-0.4907		
75.2	-0.4481		
80.2	-0.3605		
85.2	-0.2599		
90.1	-0.1521		
95.2	-0.0264		
100.0	0.0944		

	UPPER	LOWER	TOTAL
CN	0.6425	-0.0149	0.6275
CC	-0.0220	0.0171	-0.0049
CM	-0.2274	-0.0465	-0.2739

AIRFOIL PERFORMANCE

CL	CD	CM
0.6273	0.0170	-0.2739

Table 5.3.

NPL SECTION ANALYSIS
9510

RUN NO. = 353

ALPHA = 1.0

MACH NO. =0.7985

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 25

UPPER SURFACE		LOWER SURFACE	
%CHORD	CP LOCAL	%CHORD	CP LOCAL
0.0	1.0966	0.0	1.1041
0.7	-0.4336	1.0	0.6086
1.0	-0.7256	1.8	0.3393
1.6	-0.9165	5.3	-0.0952
2.0	-1.0180	10.2	-0.3690
2.6	-1.0136	15.3	-0.4551
3.1	-1.0843	20.2	-0.5262
3.6	-1.0607	27.7	-0.6448
5.2	-1.0160	37.6	-0.6167
7.7	-0.9407	45.1	-0.4788
10.2	-0.8255	52.6	-0.1056
15.2	-0.7753	60.1	0.0476
20.2	-0.1920	67.5	0.1763
25.2	-0.2876	71.7	0.2375
30.2	-0.3454	85.1	0.3382
40.2	-0.4106	90.1	0.3401
45.2	-0.4758	95.1	0.2871
50.2	-0.4595	100.0	0.0764
52.7	-0.4430		
55.2	-0.4503		
57.7	-0.4444		
60.2	-0.4810		
62.7	-0.4898		
65.2	-0.5131		
67.7	-0.5409		
70.2	-0.5295		
75.2	-0.4503		
80.2	-0.3544		
85.2	-0.2500		
90.1	-0.1403		
95.2	-0.0237		
100.0	0.0698		

	UPPER	LOWER	TOTAL
CN	0.4316	-0.1144	0.3173
CC	-0.0093	0.0129	0.0036
CM	-0.1707	-0.0133	-0.1841

AIRFOIL PERFORMANCE

CL	CD	CM
0.3172	0.0091	-0.1841

Table 5.4.

NPL SECTION ANALYSIS
9510

RUN NO. = 332

ALPHA = 0.0

MACH NO. =0.809

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 48

%CHORD	UPPER SURFACE CP LOCAL		LOWER SURFACE CP LOCAL
0.0	1.1110	0.0	1.1214
0.7	-0.2629	1.0	0.4791
1.0	-0.5392	1.8	0.2032
1.6	-0.7484	5.3	-0.2329
2.0	-0.8484	10.2	-0.5251
2.6	-0.7852	15.3	-0.6096
3.1	-0.7719	20.2	-0.6747
3.6	-0.7205	27.7	-0.7753
5.2	-0.6317	37.6	-0.7916
7.7	-0.4738	45.1	-0.7916
10.2	-0.2981	52.6	-0.2252
15.2	-0.3129	60.1	-0.0148
20.2	-0.2863	67.5	0.1095
25.2	-0.3048	71.7	0.1634
30.2	-0.3122	85.1	0.2652
40.2	-0.3581	90.1	0.2745
45.2	-0.4187	95.1	0.2397
50.2	-0.4099	100.0	0.0845
52.7	-0.3950		
55.2	-0.4067		
57.7	-0.4037		
60.2	-0.4402		
62.7	-0.4460		
65.2	-0.4764		
67.7	-0.5064		
70.2	-0.4998		
75.2	-0.4351		
80.2	-0.3448		
85.2	-0.2442		
90.1	-0.1365		
95.2	-0.0189		
100.0	0.0761		

	UPPER	LOWER	TOTAL
CN	0.3435	-0.2350	0.1084
CC	0.0001	0.0134	0.0135
CM	-0.1532	0.0358	-0.1174

AIRFOIL PERFORMANCE

CL	CD	CM
0.1084	0.0135	-0.1174

Table 5.5.

NPL SECTION ANALYSIS
9510

RUN NO. = 329

ALPHA = 0.0

MACH NO. =0.7391

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 45

UPPER SURFACE %CHORD	CP LOCAL	LOWER SURFACE %CHORD	CP LOCAL
0.0	1.0915	0.0	1.0998
0.7	-0.3841	1.0	0.4288
1.0	-0.6762	1.8	0.1473
1.6	-0.8690	5.3	-0.2930
2.0	-0.8641	10.2	-0.5434
2.6	-0.7144	15.3	-0.6022
3.1	-0.7225	20.2	-0.6577
3.6	-0.6558	27.7	-0.7328
5.2	-0.5443	37.6	-0.6495
7.7	-0.4090	45.1	-0.5500
10.2	-0.3357	52.6	-0.1798
15.2	-0.3243	60.1	-0.0098
20.2	-0.2950	67.5	0.1294
25.2	-0.3052	71.7	0.1934
30.2	-0.3101	85.1	0.3070
40.2	-0.3460	90.1	0.3101
45.2	-0.3933	95.1	0.2618
50.2	-0.3803	100.0	0.0714
52.7	-0.3679		
55.2	-0.3760		
57.7	-0.3727		
60.2	-0.4002		
62.7	-0.4082		
65.2	-0.4292		
67.7	-0.4502		
70.2	-0.4483		
75.2	-0.4152		
80.2	-0.3505		
85.2	-0.2641		
90.1	-0.1644		
95.2	-0.0450		
100.0	0.0633		

	UPPER	LOWER	TOTAL
CN	0.3343	-0.1907	0.1437
CC	-0.0009	0.0069	0.0060
CM	-0.1494	0.0119	-0.1374

AIRFOIL PERFORMANCE

CL	CD	CM
0.1437	0.0060	-0.1374

Table 5.6.

NPL SECTION ANALYSIS
9510

RUN NO. = 313

ALPHA = 4.0

MACH NO. =.6995

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 29

%CHORD	UPPER SURFACE CP LOCAL		LOWER SURFACE CP LOCAL
0.0	0.8498	0.0	0.8414
0.7	-1.4227	1.0	0.9388
1.0	-1.6925	1.8	0.7280
1.6	-1.8283	5.3	0.3348
2.0	-1.8283	10.2	0.0833
2.6	-1.8283	15.3	-0.0230
3.1	-1.8283	20.2	-0.1042
3.6	-1.8283	27.7	-0.1960
5.2	-1.8192	37.6	-0.2172
7.7	-1.8192	45.1	-0.1572
10.2	-1.6473	52.6	-0.0389
15.2	-1.6734	60.1	0.1185
20.2	-1.0849	67.5	0.2221
25.2	-0.6319	71.7	0.2750
30.2	-0.4453	85.1	0.3579
40.2	-0.4524	90.1	0.3527
45.2	-0.4981	95.1	0.2905
50.2	-0.4735	100.0	0.0501
52.7	-0.4545		
55.2	-0.4527		
57.7	-0.4336		
60.2	-0.4458		
62.7	-0.4353		
65.2	-0.4382		
67.7	-0.4514		
70.2	-0.4372		
75.2	-0.3916		
80.2	-0.3254		
85.2	-0.2428		
90.1	-0.1510		
95.2	-0.0465		
100.0	0.0472		

	UPPER	LOWER	TOTAL
CN	0.6424	0.1002	0.7426
CC	-0.0443	0.0246	-0.0197
CM	-0.1944	-0.0721	-0.2665

AIRFOIL PERFORMANCE

CL	CD	CM
0.7421	0.0321	-0.2665

Table 5.7.

NPL SECTION ANALYSIS
9510

RUN NO. = 302

ALPHA = 3.0

MACH NO. =0.7002

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 18

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9219	0.0	0.9205
0.7	-1.2646	1.0	0.8570
1.0	-1.5560	1.8	0.6242
1.6	-1.7146	5.3	0.2187
2.0	-1.8103	10.2	-0.0317
2.6	-1.7703	15.3	-0.1320
3.1	-1.8468	20.2	-0.2078
3.6	-1.8468	27.7	-0.2870
5.2	-1.7623	37.6	-0.2958
7.7	-1.6373	45.1	-0.2289
10.2	-1.5001	52.6	-0.0389
15.2	-0.9341	60.1	0.0831
20.2	-0.4428	67.5	0.1923
25.2	-0.4448	71.7	0.2486
30.2	-0.4623	85.1	0.3412
40.2	-0.4729	90.1	0.3385
45.2	-0.5116	95.1	0.2791
50.2	-0.4834	100.0	0.0515
52.7	-0.4668		
55.2	-0.4615		
57.7	-0.4406		
60.2	-0.4581		
62.7	-0.4493		
65.2	-0.4565		
67.7	-0.4700		
70.2	-0.4587		
75.2	-0.4132		
80.2	-0.3428		
85.2	-0.2567		
90.1	-0.1607		
95.2	-0.0507		
100.0	0.0444		

	UPPER	LOWER	TOTAL
CN	0.5607	0.0455	0.6062
CC	-0.0350	0.0219	-0.0131
CM	-0.1849	-0.0549	-0.2399

AIRFOIL PERFORMANCE

CL	CD	CM
0.6061	0.0186	-0.2399

Table 5.8

NPL SECTION ANALYSIS
9510

RUN NO. = 296

ALPHA = 2.0

MACH NO. =.7022

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 12

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9893	0.0	0.9895
0.7	-1.0442	1.0	0.7505
1.0	-1.3527	1.8	0.4956
1.6	-1.5326	5.3	0.0779
2.0	-1.6338	10.2	-0.1629
2.6	-1.5675	15.3	-0.2406
3.1	-1.6355	20.2	-0.3074
3.6	-1.5378	27.7	-0.3724
5.2	-1.4802	37.6	-0.3618
7.7	-1.4453	45.1	-0.2898
10.2	-0.6109	52.6	-0.0600
15.2	-0.4538	60.1	0.0600
20.2	-0.4346	67.5	0.2394
25.2	-0.4464	71.7	0.2984
30.2	-0.4376	85.1	0.3972
40.2	-0.4341	90.1	0.3990
45.2	-0.4711	95.1	0.3426
50.2	-0.4482	100.0	0.1285
52.7	-0.4259		
55.2	-0.4242		
57.7	-0.4085		
60.2	-0.4329		
62.7	-0.4294		
65.2	-0.3636		
67.7	-0.3790		
70.2	-0.3690		
75.2	-0.3276		
80.2	-0.2605		
85.2	-0.1763		
90.1	-0.0807		
95.2	0.0193		
100.0	0.1193		

	UPPER	LOWER	TOTAL
CN	0.4415	0.0173	0.4588
CC	-0.0270	0.0162	-0.0108
CM	-0.1472	-0.0604	-0.2077

AIRFOIL PERFORMANCE

CL	CD	CM
0.4589	0.0052	-0.2077

Table 5.9.

NPL SECTION ANALYSIS
9510

RUN NO. = 280

ALPHA = 2.0

MACH NO. =0.7016

WING DATA FILE NAME = *NPL11.DAT
INPUT FILE NO. - 3\3\40

%CHORD	UPPER SURFACE CP LOCAL		LOWER SURFACE CP LOCAL
0.0	0.9937	0.0	0.9955
0.7	-1.0254	1.0	0.7440
1.0	-1.3367	1.8	0.4889
1.6	-1.5287	5.3	0.0686
2.0	-1.6312	10.2	-0.1741
2.6	-1.5443	15.3	-0.2577
3.1	-1.5929	20.2	-0.3209
3.6	-1.5096	27.7	-0.3875
5.2	-1.4834	37.6	-0.3805
7.7	-1.4275	45.1	-0.3068
10.2	-0.5305	52.6	-0.0721
15.2	-0.4660	60.1	0.0510
20.2	-0.4433	67.5	0.1723
25.2	-0.4495	71.7	0.2312
30.2	-0.4425	85.1	0.3312
40.2	-0.4320	90.1	0.3308
45.2	-0.4671	95.1	0.2739
50.2	-0.4443	100.0	0.0562
52.7	-0.4249		
55.2	-0.4284		
57.7	-0.4128		
60.2	-0.4319		
62.7	-0.4301		
65.2	-0.4363		
67.7	-0.4511		
70.2	-0.4425		
75.2	-0.4013		
80.2	-0.3331		
85.2	-0.2458		
90.1	-0.1492		
95.2	-0.0400		
100.0	0.0526		

	UPPER	LOWER	TOTAL
CN	0.4636	-0.0157	0.4479
CC	-0.0238	0.0178	-0.0060
CM	-0.1674	-0.0376	-0.2049

AIRFOIL PERFORMANCE

CL	CD	CM
0.4478	0.0097	-0.2049

Table 5.10.

NPL SECTION ANALYSIS
9510

RUN NO. = 294

ALPHA = 1.0

MACH NO. =0.6993

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 10

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0484	0.0	1.0520
0.7	-0.7504	1.0	0.5907
1.0	-1.0803	1.8	0.3175
1.6	-1.2741	5.3	-0.1100
2.0	-1.2513	10.2	-0.3371
2.6	-1.1054	15.3	-0.4038
3.1	-1.1564	20.2	-0.4552
3.6	-1.1159	27.7	-0.5012
5.2	-1.0575	37.6	-0.4729
7.7	-0.4934	45.1	-0.3985
10.2	-0.4509	52.6	-0.1279
15.2	-0.4191	60.1	0.0178
20.2	-0.3749	67.5	0.1490
25.2	-0.3737	71.7	0.2106
30.2	-0.3719	85.1	0.3167
40.2	-0.3773	90.1	0.3184
45.2	-0.4162	95.1	0.2681
50.2	-0.3985	100.0	0.0654
52.7	-0.3884		
55.2	-0.3919		
57.7	-0.3814		
60.2	-0.4007		
62.7	-0.4024		
65.2	-0.4128		
67.7	-0.4285		
70.2	-0.4234		
75.2	-0.3905		
80.2	-0.3289		
85.2	-0.2465		
90.1	-0.1519		
95.2	-0.0405		
100.0	0.0583		

	UPPER	LOWER	TOTAL
CN	0.3842	-0.0884	0.2958
CC	-0.0124	0.0126	0.0002
CM	-0.1528	-0.0167	-0.1695

AIRFOIL PERFORMANCE

CL	CD	CM
0.2957	0.0054	-0.1695

Table 5.11.

NPL SECTION ANALYSIS
9510

RUN NO. = 275

ALPHA = 0.0

MACH NO. =.7016

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 35

%CHORD	UPPER SURFACE CP LOCAL	%CHORD	LOWER SURFACE CP LOCAL
0.0	1.0741	0.0	1.0796
0.7	-0.4811	1.0	0.4311
1.0	-0.7811	1.8	0.1520
1.6	-0.8980	5.3	-0.2703
2.0	-0.8557	10.2	-0.4806
2.6	-0.7110	15.3	-0.5335
3.1	-0.7145	20.2	-0.5743
3.6	-0.6475	27.7	-0.5956
5.2	-0.5301	37.6	-0.5513
7.7	-0.4046	45.1	-0.4768
10.2	-0.3304	52.6	-0.1778
15.2	-0.3145	60.1	-0.0160
20.2	-0.2880	67.5	0.1262
25.2	-0.2951	71.7	0.1893
30.2	-0.3039	85.1	0.3026
40.2	-0.3322	90.1	0.3071
45.2	-0.3746	95.1	0.2597
50.2	-0.3640	100.0	0.0669
52.7	-0.3525		
55.2	-0.3647		
57.7	-0.3578		
60.2	-0.3804		
62.7	-0.3822		
65.2	-0.4017		
67.7	-0.4196		
70.2	-0.4181		
75.2	-0.3924		
80.2	-0.3374		
85.2	-0.2595		
90.1	-0.1658		
95.2	-0.0494		
100.0	0.0635		

	UPPER	LOWER	TOTAL
CN	0.3238	-0.1537	0.1702
CC	-0.0022	0.0074	0.0052
CM	-0.1434	0.0023	-0.1412

AIRFOIL PERFORMANCE

CL	CD	CM
0.1702	0.0052	-0.1412

Table 5.12.

NPL SECTION ANALYSIS
9510

RUN NO. = 316

ALPHA = 4.0

MACH NO. =0.5988

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 32

%CHORD	UPPER SURFACE CP LOCAL	%CHORD	LOWER SURFACE CP LOCAL
0.0	0.7146	0.0	0.7027
0.7	-2.0198	1.0	0.9355
1.0	-1.9534	1.8	0.7293
1.6	-1.8788	5.3	0.3325
2.0	-1.8592	10.2	0.0842
2.6	-1.8134	15.3	-0.0221
3.1	-1.8722	20.2	-0.0996
3.6	-1.9288	27.7	-0.1748
5.2	-1.7835	37.6	-0.1991
7.7	-1.4972	45.1	-0.1460
10.2	-1.1956	52.6	-0.0554
15.2	-0.8918	60.1	0.1020
20.2	-0.6666	67.5	0.1978
25.2	-0.5795	71.7	0.2497
30.2	-0.5288	85.1	0.3275
40.2	-0.4737	90.1	0.3219
45.2	-0.4891	95.1	0.2609
50.2	-0.4605	100.0	0.0242
52.7	-0.4477		
55.2	-0.4433		
57.7	-0.4191		
60.2	-0.4301		
62.7	-0.4191		
65.2	-0.4120		
67.7	-0.4180		
70.2	-0.4038		
75.2	-0.3628		
80.2	-0.3021		
85.2	-0.2291		
90.1	-0.1488		
95.2	-0.0604		
100.0	0.0151		

	UPPER	LOWER	TOTAL
CN	0.5610	0.0915	0.6525
CC	-0.0452	0.0241	-0.0211
CM	-0.1792	-0.0639	-0.2431

AIRFOIL PERFORMANCE

CL	CD	CM
0.6524	0.0245	-0.2431

Table 5.13

NPL SECTION ANALYSIS
9510

RUN NO. = 318

ALPHA = 4.0

MACH NO. =0.5577

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 34

%CHORD	UPPER SURFACE CP LOCAL	%CHORD	LOWER SURFACE CP LOCAL
0.0	0.6301	0.0	0.6299
0.7	-2.1553	1.0	0.9350
1.0	-2.1303	1.8	0.7332
1.6	-2.0670	5.3	0.3346
2.0	-2.0327	10.2	0.0861
2.6	-2.0155	15.3	-0.0171
3.1	-2.0646	20.2	-0.0955
3.6	-2.0719	27.7	-0.1764
5.2	-1.7684	37.6	-0.2008
7.7	-1.3937	45.1	-0.1494
10.2	-1.0336	52.6	-0.0661
15.2	-0.7617	60.1	0.0931
20.2	-0.6270	67.5	0.1863
25.2	-0.5782	71.7	0.2348
30.2	-0.5389	85.1	0.3131
40.2	-0.4896	90.1	0.3071
45.2	-0.4995	95.1	0.2487
50.2	-0.4749	100.0	0.0198
52.7	-0.4605		
55.2	-0.4580		
57.7	-0.4360		
60.2	-0.4507		
62.7	-0.4409		
65.2	-0.4251		
67.7	-0.4310		
70.2	-0.4180		
75.2	-0.3772		
80.2	-0.3185		
85.2	-0.2448		
90.1	-0.1612		
95.2	-0.0694		
100.0	0.0119		

	UPPER	LOWER	TOTAL
CN	0.5604	0.0851	0.6456
CC	-0.0469	0.0242	-0.0226
CM	-0.1833	-0.0592	-0.2425

AIRFOIL PERFORMANCE

CL	CD	CM
0.6456	0.0225	-0.2425

Table 5.14

NPL SECTION ANALYSIS
9510

RUN NO. = 324

ALPHA = 6.0

MACH NO. =0.504

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 40

%CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE	CP LOCAL
0.0	0.3628	0.0	0.3589
0.7	-2.0629	1.0	1.0068
1.0	-1.6708	1.8	0.8638
1.6	-2.0928	5.3	0.4961
2.0	-1.6553	10.2	0.2393
2.6	-1.9402	15.3	0.1190
3.1	-1.6754	20.2	0.0348
3.6	-1.7733	27.7	-0.0668
5.2	-1.6437	37.6	-0.1103
7.7	-1.5948	45.1	-0.0784
10.2	-1.4537	52.6	-0.0116
15.2	-1.2868	60.1	0.1335
20.2	-1.0162	67.5	0.2031
25.2	-0.8284	71.7	0.2489
30.2	-0.6812	85.1	0.3123
40.2	-0.5051	90.1	0.3005
45.2	-0.4849	95.1	0.2325
50.2	-0.4416	100.0	-0.0305
52.7	-0.4313		
55.2	-0.4197		
57.7	-0.3878		
60.2	-0.3878		
62.7	-0.3792		
65.2	-0.3598		
67.7	-0.3561		
70.2	-0.3409		
75.2	-0.3036		
80.2	-0.2534		
85.2	-0.1985		
90.1	-0.1425		
95.2	-0.0831		
100.0	-0.0386		

	UPPER	LOWER	TOTAL
CN	0.6173	0.1462	0.7635
CC	-0.0529	0.0260	-0.0269
CM	-0.1847	-0.0743	-0.2591

AIRFOIL PERFORMANCE

CL	CD	CM
0.7621	0.0531	-0.2591

Table 5.15.

NPL SECTION ANALYSIS
9510

RUN NO. = 321

ALPHA = 5.0

MACH NO. =0.5055

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 37

%CHORD	UPPER SURFACE CP LOCAL		LOWER SURFACE CP LOCAL
0.0	0.4616	0.0	0.4636
0.7	-2.1310	1.0	0.9763
1.0	-1.7826	1.8	0.8063
1.6	-2.0995	5.3	0.4262
2.0	-1.7491	10.2	0.1728
2.6	-1.9932	15.3	0.0547
3.1	-1.7720	20.2	-0.0259
3.6	-1.9243	27.7	-0.1152
5.2	-1.7634	37.6	-0.1469
7.7	-1.5394	45.1	-0.1037
10.2	-1.2407	52.6	-0.0405
15.2	-0.9277	60.1	0.1158
20.2	-0.6950	67.5	0.2004
25.2	-0.5862	71.7	0.2458
30.2	-0.5284	85.1	0.3180
40.2	-0.4562	90.1	0.3104
45.2	-0.4591	95.1	0.2471
50.2	-0.4331	100.0	0.0141
52.7	-0.4233		
55.2	-0.4118		
57.7	-0.3974		
60.2	-0.4032		
62.7	-0.3917		
65.2	-0.3764		
67.7	-0.3773		
70.2	-0.3644		
75.2	-0.3247		
80.2	-0.2722		
85.2	-0.2079		
90.1	-0.1378		
95.2	-0.0594		
100.0	0.0058		

	UPPER	LOWER	TOTAL
CN	0.5532	0.1216	0.6748
CC	-0.0494	0.0250	-0.0244
CM	-0.1704	-0.0697	-0.2401

AIRFOIL PERFORMANCE

CL	CD	CM
0.6743	0.0345	-0.2401

Table 5.16.

NPL SECTION ANALYSIS
9510

RUN NO. = 306

ALPHA = 4.0

MACH NO. =0.5029

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 22

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.5426	0.0	0.5383
0.7	-2.3952	1.0	0.9420
1.0	-2.2405	1.8	0.7489
1.6	-2.2908	5.3	0.3510
2.0	-2.1498	10.2	0.1053
2.6	-2.2275	15.3	-0.0059
3.1	-2.1815	20.2	-0.0819
3.6	-2.0664	27.7	-0.1580
5.2	-1.4989	37.6	-0.1814
7.7	-1.0156	45.1	-0.1316
10.2	-0.7928	52.6	-0.0614
15.2	-0.6713	60.1	0.1024
20.2	-0.5758	67.5	0.1941
25.2	-0.5353	71.7	0.2426
30.2	-0.4946	85.1	0.3207
40.2	-0.4480	90.1	0.3160
45.2	-0.4539	95.1	0.2580
50.2	-0.4335	100.0	0.0438
52.7	-0.4219		
55.2	-0.4190		
57.7	-0.4015		
60.2	-0.4131		
62.7	-0.4044		
65.2	-0.3874		
67.7	-0.3943		
70.2	-0.3825		
75.2	-0.3437		
80.2	-0.2895		
85.2	-0.2189		
90.1	-0.1389		
95.2	-0.0483		
100.0	0.0377		

	UPPER	LOWER	TOTAL
CN	0.5074	0.0971	0.6045
CC	-0.0483	0.0236	-0.0248
CM	-0.1647	-0.0647	-0.2293

AIRFOIL PERFORMANCE

CL	CD	CM
0.6047	0.0175	-0.2293

Table 5.17.

NPL SECTION ANALYSIS
9510

RUN NO. = 304

ALPHA = 3.0

MACH NO. =0.5022

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 20

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.6932	0.0	0.6756
0.7	-2.2609	1.0	0.8687
1.0	-2.3457	1.8	0.6493
1.6	-2.0889	5.3	0.2398
2.0	-2.0340	10.2	0.0058
2.6	-1.9504	15.3	-0.0909
3.1	-2.0312	20.2	-0.1584
3.6	-1.6359	27.7	-0.2258
5.2	-0.9596	37.6	-0.2405
7.7	-0.8196	45.1	-0.1818
10.2	-0.6854	52.6	-0.0851
15.2	-0.5921	60.1	0.0733
20.2	-0.5163	67.5	0.1732
25.2	-0.4741	71.7	0.2230
30.2	-0.4509	85.1	0.3085
40.2	-0.4189	90.1	0.3061
45.2	-0.4276	95.1	0.2527
50.2	-0.4131	100.0	0.0486
52.7	-0.4101		
55.2	-0.4072		
57.7	-0.3869		
60.2	-0.3985		
62.7	-0.3927		
65.2	-0.3870		
67.7	-0.3930		
70.2	-0.3848		
75.2	-0.3518		
80.2	-0.3000		
85.2	-0.2306		
90.1	-0.1483		
95.2	-0.0494		
100.0	0.0435		

	UPPER	LOWER	TOTAL
CN	0.4612	0.0529	0.5141
CC	-0.0407	0.0217	-0.0189
CM	-0.1587	-0.0511	-0.2098

AIRFOIL PERFORMANCE

CL	CD	CM
0.5144	0.0080	-0.2098

Table 5.18.

NPL SECTION ANALYSIS
9510

RUN NO. = 301

ALPHA = 2.

MACH NO. =0.5028

WING DATA FILE NAME = *NPL12.DAT
INPUT FILE NO. - 17

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.8366	0.0	0.8236
0.7	-1.7025	1.0	0.7648
1.0	-1.8809	1.8	0.5206
1.6	-1.6277	5.3	0.1147
2.0	-1.5667	10.2	-0.1000
2.6	-1.4652	15.3	-0.1799
3.1	-1.5377	20.2	-0.2360
3.6	-1.2882	27.7	-0.2920
5.2	-0.7664	37.6	-0.2920
7.7	-0.6699	45.1	-0.2242
10.2	-0.5616	52.6	-0.0887
15.2	-0.4914	60.1	0.0503
20.2	-0.4300	67.5	0.1599
25.2	-0.4095	71.7	0.2136
30.2	-0.3891	85.1	0.3043
40.2	-0.3686	90.1	0.3039
45.2	-0.3774	95.1	0.2548
50.2	-0.3657	100.0	0.0598
52.7	-0.3657		
55.2	-0.3657		
57.7	-0.3540		
60.2	-0.3686		
62.7	-0.3657		
65.2	-0.3597		
67.7	-0.3692		
70.2	-0.3645		
75.2	-0.3361		
80.2	-0.2875		
85.2	-0.2213		
90.1	-0.1420		
95.2	-0.0414		
100.0	0.0556		

	UPPER	LOWER	TOTAL
CN	0.3955	0.0130	0.4085
CC	-0.0281	0.0184	-0.0097
CM	-0.1434	-0.0413	-0.1847

AIRFOIL PERFORMANCE

CL	CD	CM
0.4086	0.0046	-0.1847

Table 5.19.

NPL SECTION ANALYSIS
9510

RUN NO. = 356

ALPHA = 1.0

MACH NO. =0.4975

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 28

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9684	0.0	0.9594
0.7	-1.0307	1.0	0.5614
1.0	-1.2268	1.8	0.2970
1.6	-1.1343	5.3	-0.0921
2.0	-1.0575	10.2	-0.2644
2.6	-0.8921	15.3	-0.3081
3.1	-0.9069	20.2	-0.3436
3.6	-0.8507	27.7	-0.3673
5.2	-0.5302	37.6	-0.3525
7.7	-0.4799	45.1	-0.2814
10.2	-0.4117	52.6	-0.0918
15.2	-0.3673	60.1	0.0237
20.2	-0.3288	67.5	0.2648
25.2	-0.3199	71.7	0.3199
30.2	-0.3140	85.1	0.4182
40.2	-0.3140	90.1	0.4206
45.2	-0.3288	95.1	0.3750
50.2	-0.3229	100.0	0.1953
52.7	-0.3267		
55.2	-0.3297		
57.7	-0.3238		
60.2	-0.3386		
62.7	-0.3386		
65.2	-0.2049		
67.7	-0.2157		
70.2	-0.2133		
75.2	-0.1917		
80.2	-0.1498		
85.2	-0.0899		
90.1	-0.0156		
95.2	0.0829		
100.0	0.1886		

	UPPER	LOWER	TOTAL
CN	0.2745	-0.0002	0.2743
CC	-0.0176	0.0083	-0.0094
CM	-0.0905	-0.0642	-0.1548

AIRFOIL PERFORMANCE

CL	CD	CM
0.2744	-0.0046	-0.1548

Table 5.20.

NPL SECTION ANALYSIS
9510

RUN NO. = 288

ALPHA = 0.0

MACH NO. =0.506

WING DATA FILE NAME = *NPL11.DAT
INPUT FILE NO. - 48

%CHORD	UPPER SURFACE CP LOCAL	%CHORD	LOWER SURFACE CP LOCAL
0.0	1.0224	0.0	1.0164
0.7	-0.5933	1.0	0.3764
1.0	-0.8007	1.8	0.1129
1.6	-0.7849	5.3	-0.2490
2.0	-0.7333	10.2	-0.3909
2.6	-0.6102	15.3	-0.4170
3.1	-0.5987	20.2	-0.4344
3.6	-0.5385	27.7	-0.4459
5.2	-0.4452	37.6	-0.4170
7.7	-0.3734	45.1	-0.3446
10.2	-0.3188	52.6	-0.1332
15.2	-0.2700	60.1	-0.0058
20.2	-0.2499	67.5	0.1171
25.2	-0.2592	71.7	0.1757
30.2	-0.2621	85.1	0.2800
40.2	-0.2736	90.1	0.2846
45.2	-0.2966	95.1	0.2436
50.2	-0.2909	100.0	0.0703
52.7	-0.2930		
55.2	-0.2987		
57.7	-0.2930		
60.2	-0.3073		
62.7	-0.3073		
65.2	-0.3114		
67.7	-0.3262		
70.2	-0.3250		
75.2	-0.3087		
80.2	-0.2691		
85.2	-0.2132		
90.1	-0.1421		
95.2	-0.0442		
100.0	0.0662		

	UPPER	LOWER	TOTAL
CN	0.2702	-0.1023	0.1679
CC	-0.0036	0.0060	0.0024
CM	-0.1165	-0.0109	-0.1274

AIRFOIL PERFORMANCE

CL	CD	CM
0.1679	0.0024	-0.1274

Table 5.21

NPL SECTION ANALYSIS
9510

RUN NO. = 398

ALPHA = 2.0

MACH NO. =.8702

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

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.1098	0.0	1.1366
0.7	-0.3284	1.0	0.7246
1.0	-0.5978	1.8	0.4723
1.6	-0.8736	5.3	0.0698
2.0	-0.8789	10.2	-0.2040
2.6	-0.9245	15.3	-0.3022
3.1	-0.9782	20.2	-0.3718
3.6	-1.0024	27.7	-0.5082
5.2	-0.9548	37.6	-0.7088
7.7	-0.8782	45.1	-0.6834
10.2	-0.8311	52.6	-0.0953
15.2	-0.8486	60.1	0.0537
20.2	-0.8271	67.5	0.1585
25.2	-0.8153	71.7	0.2169
30.2	-0.8233	85.1	0.3170
40.2	-0.6645	90.1	0.3186
45.2	-0.6859	95.1	0.2611
50.2	-0.6792	100.0	0.0212
52.7	-0.6709		
55.2	-0.6946		
57.7	-0.6801		
60.2	-0.7156		
62.7	-0.6301		
65.2	-0.6126		
67.7	-0.4853		
70.2	-0.3498		
75.2	-0.2656		
80.2	-0.2170		
85.2	-0.1565		
90.1	-0.0928		
95.2	-0.0342		
100.0	0.0245		

	UPPER	LOWER	TOTAL
CN	0.5607	-0.0966	0.4641
CC	-0.0151	0.0236	0.0085
CM	-0.2062	-0.0050	-0.2112

AIRFOIL PERFORMANCE

CL	CD	CM
0.4635	0.0247	-0.2112

Table 5.22.

NPL SECTION ANALYSIS
9510

RUN NO. = 396

ALPHA = 2.0

MACH NO. =.8488

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 50

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0937	0.0	1.1312
0.7	-0.4101	1.0	0.7293
1.0	-0.6960	1.8	0.4748
1.6	-0.9689	5.3	0.0702
2.0	-0.9703	10.2	-0.1968
2.6	-1.0076	15.3	-0.2907
3.1	-1.0685	20.2	-0.3662
3.6	-1.0976	27.7	-0.5006
5.2	-1.0384	37.6	-0.6925
7.7	-0.9597	45.1	-0.2455
10.2	-0.9017	52.6	-0.0798
15.2	-0.9183	60.1	0.0592
20.2	-0.8921	67.5	0.1725
25.2	-0.8763	71.7	0.2280
30.2	-0.8790	85.1	0.3256
40.2	-0.6843	90.1	0.3262
45.2	-0.7131	95.1	0.2700
50.2	-0.7063	100.0	0.0367
52.7	-0.6951		
55.2	-0.6762		
57.7	-0.4805		
60.2	-0.3644		
62.7	-0.3158		
65.2	-0.3447		
67.7	-0.3706		
70.2	-0.3850		
75.2	-0.3556		
80.2	-0.2796		
85.2	-0.1908		
90.1	-0.1015		
95.2	-0.0240		
100.0	0.0535		

	UPPER	LOWER	TOTAL
CN	0.5675	-0.0550	0.5125
CC	-0.0182	0.0192	0.0010
CM	-0.2000	-0.0244	-0.2244

AIRFOIL PERFORMANCE

CL	CD	CM
0.5122	0.0189	-0.2244

Table 5.23.

NPL SECTION ANALYSIS
9510

RUN NO. = 395

ALPHA = 1.0

MACH NO. =.8438

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 43

UPPER SURFACE		LOWER SURFACE	
%CHORD	CP LOCAL	%CHORD	CP LOCAL
0.0	1.1101	0.0	1.1367
0.7	-0.3070	1.0	0.6299
1.0	-0.5974	1.8	0.3669
1.6	-0.8612	5.3	-0.0485
2.0	-0.8529	10.2	-0.3309
2.6	-0.9813	15.3	-0.4227
3.1	-1.0413	20.2	-0.4904
3.6	-1.0301	27.7	-0.6133
5.2	-0.9344	37.6	-0.8191
7.7	-0.8662	45.1	-0.5249
10.2	-0.7826	52.6	-0.2033
15.2	-0.7840	60.1	-0.0650
20.2	-0.7311	67.5	0.0631
25.2	-0.2793	71.7	0.1285
30.2	-0.2420	85.1	0.2648
40.2	-0.3360	90.1	0.2825
45.2	-0.4190	95.1	0.2478
50.2	-0.4107	100.0	0.0738
52.7	-0.3901		
55.2	-0.3942		
57.7	-0.3860		
60.2	-0.4268		
62.7	-0.4431		
65.2	-0.4963		
67.7	-0.5409		
70.2	-0.5790		
75.2	-0.5588		
80.2	-0.3110		
85.2	-0.2131		
90.1	-0.1113		
95.2	-0.0103		
100.0	0.0907		

	UPPER	LOWER	TOTAL
CN	0.4259	-0.1675	0.2584
CC	-0.0082	0.0219	0.0137
CM	-0.1644	0.0233	-0.1411

AIRFOIL PERFORMANCE

CL	CD	CM
0.2581	0.0182	-0.1411

Table 5.24.

NPL SECTION ANALYSIS
9510

RUN NO. = 394

ALPHA = 0.0

MACH NO. =.8375

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 40

UPPER SURFACE		LOWER SURFACE	
%CHORD	CP LOCAL	%CHORD	CP LOCAL
0.0	1.1191	0.0	1.1402
0.7	-0.2070	1.0	0.5199
1.0	-0.4686	1.8	0.2523
1.6	-0.7196	5.3	-0.1659
2.0	-0.7857	10.2	-0.4600
2.6	-0.9332	15.3	-0.5549
3.1	-0.9642	20.2	-0.6133
3.6	-0.8855	27.7	-0.7134
5.2	-0.7955	37.6	-0.9067
7.7	-0.3681	45.1	-0.4103
10.2	-0.3244	52.6	-0.3236
15.2	-0.3004	60.1	-0.2653
20.2	-0.2638	67.5	-0.1721
25.2	-0.2434	71.7	-0.1153
30.2	-0.2489	85.1	0.0754
40.2	-0.2851	90.1	0.1371
45.2	-0.3435	95.1	0.1596
50.2	-0.3324	100.0	0.0854
52.7	-0.3205		
55.2	-0.3342		
57.7	-0.3383		
60.2	-0.3794		
62.7	-0.3973		
65.2	-0.4519		
67.7	-0.4950		
70.2	-0.5356		
75.2	-0.5440		
80.2	-0.3246		
85.2	-0.2250		
90.1	-0.1153		
95.2	-0.0078		
100.0	0.0997		

	UPPER	LOWER	TOTAL
CN	0.3195	-0.2921	0.0274
CC	0.0006	0.0253	0.0258
CM	-0.1416	0.0918	-0.0498

AIRFOIL PERFORMANCE

CL	CD	CM
0.0274	0.0258	-0.0498

Table 5.25.

NPL SECTION ANALYSIS
9510

RUN NO. = 389

ALPHA = 3.0

MACH NO. =.8038

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 21

UPPER SURFACE %CHORD	CP LOCAL	LOWER SURFACE %CHORD	CP LOCAL
0.0	1.0227	0.0	1.0684
0.7	-0.7242	1.0	0.8331
1.0	-1.0257	1.8	0.5949
1.6	-1.2861	5.3	0.1998
2.0	-1.2979	10.2	-0.0488
2.6	-1.3172	15.3	-0.1493
3.1	-1.3824	20.2	-0.2291
3.6	-1.3838	27.7	-0.3163
5.2	-1.3318	37.6	-0.3310
7.7	-1.2563	45.1	-0.2158
10.2	-1.1720	52.6	-0.0488
15.2	-1.1809	60.1	0.0888
20.2	-1.1601	67.5	0.2283
25.2	-1.1279	71.7	0.2789
30.2	-1.1249	85.1	0.3650
40.2	-1.0382	90.1	0.3590
45.2	-0.7661	95.1	0.2985
50.2	-0.5044	100.0	0.0471
52.7	-0.4348		
55.2	-0.3941		
57.7	-0.3548		
60.2	-0.3606		
62.7	-0.3504		
65.2	-0.3504		
67.7	-0.3618		
70.2	-0.3576		
75.2	-0.3201		
80.2	-0.2577		
85.2	-0.1816		
90.1	-0.0983		
95.2	-0.0215		
100.0	0.0554		

	UPPER	LOWER	TOTAL
CN	0.6622	0.0454	0.7077
CC	-0.0318	0.0213	-0.0105
CM	-0.2118	-0.0597	-0.2715

AIRFOIL PERFORMANCE

CL	CD	CM
0.7073	0.0266	-0.2715

Table 5.26.

NPL SECTION ANALYSIS
9510

RUN NO. = 388

ALPHA = 2.0

MACH NO. =.8019

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 16

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0603	0.0	1.0953
0.7	-0.5789	1.0	0.7327
1.0	-0.8846	1.8	0.4800
1.6	-1.1874	5.3	0.0684
2.0	-1.1726	10.2	-0.1873
2.6	-1.1874	15.3	-0.2772
3.1	-1.2716	20.2	-0.3498
3.6	-1.2894	27.7	-0.4506
5.2	-1.2199	37.6	-0.4284
7.7	-1.1252	45.1	-0.2787
10.2	-1.0572	52.6	-0.0998
15.2	-1.0587	60.1	0.0536
20.2	-1.0203	67.5	0.1721
25.2	-0.9774	71.7	0.2273
30.2	-0.4628	85.1	0.3250
40.2	-0.3238	90.1	0.3261
45.2	-0.3978	95.1	0.2760
50.2	-0.4007	100.0	0.0627
52.7	-0.3923		
55.2	-0.4040		
57.7	-0.3981		
60.2	-0.4273		
62.7	-0.4302		
65.2	-0.4743		
67.7	-0.4941		
70.2	-0.4845		
75.2	-0.4259		
80.2	-0.3397		
85.2	-0.2422		
90.1	-0.1388		
95.2	-0.0290		
100.0	0.0808		

	UPPER	LOWER	TOTAL
CN	0.5306	-0.0293	0.5013
CC	-0.0207	0.0186	-0.0021
CM	-0.1809	-0.0332	-0.2140

AIRFOIL PERFORMANCE

CL	CD	CM
0.5011	0.0154	-0.2140

Table 5.27.

NPL SECTION ANALYSIS
9510

RUN NO. = 387

ALPHA = 1.0

MACH NO. =.8018

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 12

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0922	0.0	1.1192
0.7	-0.4271	1.0	0.6004
1.0	-0.7306	1.8	0.3350
1.6	-0.9967	5.3	-0.0889
2.0	-1.0087	10.2	-0.3543
2.6	-1.1174	15.3	-0.4297
3.1	-1.2172	20.2	-0.4918
3.6	-1.1651	27.7	-0.6158
5.2	-1.0742	37.6	-0.7517
7.7	-0.9640	45.1	-0.3293
10.2	-0.7837	52.6	-0.1406
15.2	-0.3561	60.1	0.0163
20.2	-0.3218	67.5	0.1344
25.2	-0.3205	71.7	0.1906
30.2	-0.3278	85.1	0.2951
40.2	-0.3544	90.1	0.3030
45.2	-0.4061	95.1	0.2611
50.2	-0.3869	100.0	0.0719
52.7	-0.3767		
55.2	-0.3855		
57.7	-0.3811		
60.2	-0.4132		
62.7	-0.4176		
65.2	-0.4532		
67.7	-0.4749		
70.2	-0.4719		
75.2	-0.4205		
80.2	-0.3369		
85.2	-0.2395		
90.1	-0.1320		
95.2	-0.0208		
100.0	0.0904		

	UPPER	LOWER	TOTAL
CN	0.3923	-0.1268	0.2656
CC	-0.0095	0.0148	0.0053
CM	-0.1526	-0.0017	-0.1542

AIRFOIL PERFORMANCE

CL	CD	CM
0.2654	0.0099	-0.1542

Table 5.28.

NPL SECTION ANALYSIS
9510

RUN NO. = 386

ALPHA = 0.0

MACH NO. =.8012

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 8

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.1070	0.0	1.1277
0.7	-0.2727	1.0	0.4719
1.0	-0.5394	1.8	0.1988
1.6	-0.8043	5.3	-0.2285
2.0	-0.8997	10.2	-0.5149
2.6	-0.8446	15.3	-0.5824
3.1	-0.8922	20.2	-0.6343
3.6	-0.8997	27.7	-0.7306
5.2	-0.4336	37.6	-0.9336
7.7	-0.3992	45.1	-0.3735
10.2	-0.3439	52.6	-0.1721
15.2	-0.3050	60.1	-0.0267
20.2	-0.2676	67.5	0.0887
25.2	-0.2528	71.7	0.1451
30.2	-0.2587	85.1	0.2620
40.2	-0.2883	90.1	0.2761
45.2	-0.3444	95.1	0.2437
50.2	-0.3326	100.0	0.0798
52.7	-0.3204		
55.2	-0.3320		
57.7	-0.3320		
60.2	-0.3655		
62.7	-0.3728		
65.2	-0.4240		
67.7	-0.4491		
70.2	-0.4527		
75.2	-0.4126		
80.2	-0.3337		
85.2	-0.2392		
90.1	-0.1333		
95.2	-0.0181		
100.0	0.0972		

	UPPER	LOWER	TOTAL
CN	0.3066	-0.2075	0.0990
CC	-0.0001	0.0107	0.0106
CM	-0.1352	0.0251	-0.1101

AIRFOIL PERFORMANCE

CL	CD	CM
0.0990	0.0106	-0.1101

Table 5.29.

NPL SECTION ANALYSIS
9510

RUN NO. = 393

ALPHA = 3.0

MACH NO. =.7582

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 35

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9732	0.0	1.0178
0.7	-0.9621	1.0	0.8521
1.0	-1.2770	1.8	0.6170
1.6	-1.5165	5.3	0.2257
2.0	-1.5197	10.2	-0.0158
2.6	-1.5575	15.3	-0.1168
3.1	-1.6064	20.2	-0.1909
3.6	-1.6128	27.7	-0.2730
5.2	-1.5517	37.6	-0.2872
7.7	-1.4714	45.1	-0.1988
10.2	-1.3628	52.6	-0.0457
15.2	-1.3581	60.1	0.0898
20.2	-1.3204	67.5	0.1966
25.2	-1.0152	71.7	0.2515
30.2	-0.6255	85.1	0.3383
40.2	-0.4008	90.1	0.3328
45.2	-0.4290	95.1	0.2747
50.2	-0.4102	100.0	0.0491
52.7	-0.3995		
55.2	-0.4042		
57.7	-0.3918		
60.2	-0.4166		
62.7	-0.4135		
65.2	-0.4348		
67.7	-0.4444		
70.2	-0.4361		
75.2	-0.3910		
80.2	-0.3217		
85.2	-0.2378		
90.1	-0.1430		
95.2	-0.0391		
100.0	0.0648		

	UPPER	LOWER	TOTAL
CN	0.6172	0.0526	0.6697
CC	-0.0345	0.0226	-0.0119
CM	-0.1921	-0.0566	-0.2487

AIRFOIL PERFORMANCE

CL	CD	CM
0.6694	0.0232	-0.2487

Table 5.30.

NPL SECTION ANALYSIS
9510

RUN NO. = 391

ALPHA = 2.0

MACH NO. =.7494

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 28

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0225	0.0	1.0578
0.7	-0.7983	1.0	0.7397
1.0	-1.1370	1.8	0.4877
1.6	-1.4560	5.3	0.0791
2.0	-1.4333	10.2	-0.1583
2.6	-1.4430	15.3	-0.2348
3.1	-1.5370	20.2	-0.3007
3.6	-1.5370	27.7	-0.3715
5.2	-1.4519	37.6	-0.3651
7.7	-1.3291	45.1	-0.2525
10.2	-1.2226	52.6	-0.0937
15.2	-0.6105	60.1	0.0517
20.2	-0.4199	67.5	0.1685
25.2	-0.3983	71.7	0.2218
30.2	-0.4048	85.1	0.3155
40.2	-0.4129	90.1	0.3182
45.2	-0.4564	95.1	0.2665
50.2	-0.4306	100.0	0.0608
52.7	-0.4098		
55.2	-0.4129		
57.7	-0.3970		
60.2	-0.4193		
62.7	-0.4161		
65.2	-0.4455		
67.7	-0.4573		
70.2	-0.4488		
75.2	-0.4032		
80.2	-0.3308		
85.2	-0.2414		
90.1	-0.1416		
95.2	-0.0306		
100.0	0.0803		

	UPPER	LOWER	TOTAL
CN	0.4776	-0.0100	0.4676
CC	-0.0229	0.0188	-0.0041
CM	-0.1658	-0.0372	-0.2030

AIRFOIL PERFORMANCE

CL	CD	CM
0.4675	0.0123	-0.2030

Table 5.31.

NPL SECTION ANALYSIS
9510

RUN NO. = 403

ALPHA = 0.0

MACH NO. =.7526

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 18

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0892	0.0	1.1052
0.7	-0.3657	1.0	0.4456
1.0	-0.6615	1.8	0.1681
1.6	-0.9548	5.3	-0.2632
2.0	-0.9990	10.2	-0.5090
2.6	-0.8450	15.3	-0.5598
3.1	-0.8663	20.2	-0.6088
3.6	-0.7386	27.7	-0.6847
5.2	-0.5220	37.6	-0.5598
7.7	-0.4708	45.1	-0.3811
10.2	-0.4014	52.6	-0.1839
15.2	-0.3667	60.1	-0.0206
20.2	-0.3353	67.5	0.0992
25.2	-0.2681	71.7	0.1561
30.2	-0.2712	85.1	0.2661
40.2	-0.2965	90.1	0.2765
45.2	-0.3454	95.1	0.2367
50.2	-0.3312	100.0	0.0661
52.7	-0.3154		
55.2	-0.3262		
57.7	-0.3247		
60.2	-0.3527		
62.7	-0.3558		
65.2	-0.4163		
67.7	-0.4217		
70.2	-0.4223		
75.2	-0.3942		
80.2	-0.3343		
85.2	-0.2513		
90.1	-0.1531		
95.2	-0.0351		
100.0	0.0830		

	UPPER	LOWER	TOTAL
CN	0.3203	-0.1703	0.1500
CC	-0.0025	0.0077	0.0052
CM	-0.1373	0.0112	-0.1261

AIRFOIL PERFORMANCE

CL	CD	CM
0.1500	0.0052	-0.1261

Table 5.32.

NPL SECTION ANALYSIS
9510

RUN NO. = 402

ALPHA = 0.0

MACH NO. =.7426

WING DATA FILE NAME = *NPL16.DAT
INPUT FILE NO. - 16

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0843	0.0	1.1021
0.7	-0.3861	1.0	0.4418
1.0	-0.6789	1.8	0.1639
1.6	-1.0120	5.3	-0.2651
2.0	-1.0323	10.2	-0.5044
2.6	-0.8768	15.3	-0.5551
3.1	-0.8887	20.2	-0.5984
3.6	-0.7535	27.7	-0.6433
5.2	-0.5599	37.6	-0.5503
7.7	-0.5036	45.1	-0.3754
10.2	-0.4302	52.6	-0.1765
15.2	-0.3960	60.1	-0.0144
20.2	-0.3602	67.5	0.1684
25.2	-0.2652	71.7	0.2261
30.2	-0.2684	85.1	0.3339
40.2	-0.2908	90.1	0.3423
45.2	-0.3355	95.1	0.3040
50.2	-0.3227	100.0	0.1359
52.7	-0.3144		
55.2	-0.3223		
57.7	-0.3191		
60.2	-0.3492		
62.7	-0.3507		
65.2	-0.3507		
67.7	-0.3451		
70.2	-0.3471		
75.2	-0.3206		
80.2	-0.2624		
85.2	-0.1810		
90.1	-0.0841		
95.2	0.0330		
100.0	0.1502		

	UPPER	LOWER	TOTAL
CN	0.2991	-0.1388	0.1603
CC	-0.0061	0.0056	-0.0005
CM	-0.1163	-0.0113	-0.1276

AIRFOIL PERFORMANCE

CL	CD	CM
0.1603	-0.0005	-0.1276

Table 5.33.

NPL SECTION ANALYSIS
9510

RUN NO. = 384

ALPHA = 4.0

MACH NO. =.6962

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 48

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.8573	0.0	0.9077
0.7	-1.4341	1.0	0.9238
1.0	-1.7163	1.8	0.7080
1.6	-1.9285	5.3	0.3228
2.0	-1.9905	10.2	0.0802
2.6	-1.9746	15.3	-0.0287
3.1	-2.0064	20.2	-0.1003
3.6	-2.0064	27.7	-0.1846
5.2	-1.9985	37.6	-0.2079
7.7	-1.8956	45.1	-0.1451
10.2	-1.6560	52.6	-0.0072
15.2	-1.0294	60.1	0.1073
20.2	-0.7916	67.5	0.2029
25.2	-0.6666	71.7	0.2537
30.2	-0.5937	85.1	0.3312
40.2	-0.4906	90.1	0.3220
45.2	-0.5031	95.1	0.2620
50.2	-0.4551	100.0	0.0268
52.7	-0.4248		
55.2	-0.4161		
57.7	-0.3932		
60.2	-0.4038		
62.7	-0.3932		
65.2	-0.4042		
67.7	-0.4063		
70.2	-0.3912		
75.2	-0.3459		
80.2	-0.2855		
85.2	-0.2143		
90.1	-0.1388		
95.2	-0.0612		
100.0	0.0163		

	UPPER	LOWER	TOTAL
CN	0.6083	0.0945	0.7029
CC	-0.0446	0.0247	-0.0198
CM	-0.1828	-0.0663	-0.2491

AIRFOIL PERFORMANCE

CL	CD	CM
0.7026	0.0292	-0.2491

Table 5.34.

NPL SECTION ANALYSIS
9510

RUN NO. = 383

ALPHA = 3.0

MACH NO. =.6963

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 43

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9105	0.0	0.9597
0.7	-1.2797	1.0	0.8582
1.0	-1.5957	1.8	0.6268
1.6	-1.8577	5.3	0.2333
2.0	-1.8631	10.2	-0.0036
2.6	-1.8915	15.3	-0.1018
3.1	-1.9822	20.2	-0.1697
3.6	-1.9431	27.7	-0.2411
5.2	-1.8511	37.6	-0.2536
7.7	-1.4766	45.1	-0.1750
10.2	-1.2056	52.6	-0.0410
15.2	-0.8096	60.1	0.0856
20.2	-0.5671	67.5	0.1871
25.2	-0.4970	71.7	0.2399
30.2	-0.4704	85.1	0.3251
40.2	-0.4384	90.1	0.3210
45.2	-0.4686	95.1	0.2655
50.2	-0.4384	100.0	0.0469
52.7	-0.4154		
55.2	-0.4154		
57.7	-0.3909		
60.2	-0.4067		
62.7	-0.3979		
65.2	-0.4155		
67.7	-0.4215		
70.2	-0.4114		
75.2	-0.3662		
80.2	-0.3023		
85.2	-0.2240		
90.1	-0.1364		
95.2	-0.0374		
100.0	0.0617		

	UPPER	LOWER	TOTAL
CN	0.5293	0.0592	0.5885
CC	-0.0362	0.0226	-0.0136
CM	-0.1688	-0.0566	-0.2253

AIRFOIL PERFORMANCE

CL	CD	CM
0.5884	0.0172	-0.2253

Table 5.35.

NPL SECTION ANALYSIS
9510

RUN NO. = 382

ALPHA = 2.0

MACH NO. =.701

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 38

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9893	0.0	1.0301
0.7	-1.0087	1.0	0.7285
1.0	-1.3638	1.8	0.4762
1.6	-1.6845	5.3	0.0688
2.0	-1.6473	10.2	-0.1605
2.6	-1.6615	15.3	-0.2335
3.1	-1.7906	20.2	-0.2932
3.6	-1.6668	27.7	-0.3511
5.2	-1.4987	37.6	-0.3459
7.7	-0.8953	45.1	-0.2458
10.2	-0.6211	52.6	-0.0932
15.2	-0.5131	60.1	0.0475
20.2	-0.4547	67.5	0.2109
25.2	-0.4269	71.7	0.2656
30.2	-0.4145	85.1	0.3572
40.2	-0.3969	90.1	0.3571
45.2	-0.4304	95.1	0.3080
50.2	-0.4039	100.0	0.1117
52.7	-0.3833		
55.2	-0.3850		
57.7	-0.3728		
60.2	-0.3920		
62.7	-0.3885		
65.2	-0.3550		
67.7	-0.3646		
70.2	-0.3582		
75.2	-0.3218		
80.2	-0.2590		
85.2	-0.1798		
90.1	-0.0878		
95.2	0.0202		
100.0	0.1283		

	UPPER	LOWER	TOTAL
CN	0.4205	0.0086	0.4291
CC	-0.0261	0.0170	-0.0091
CM	-0.1397	-0.0511	-0.1907

AIRFOIL PERFORMANCE

CL	CD	CM
0.4292	0.0059	-0.1907

Table 5.36.

NPL SECTION ANALYSIS
9510

RUN NO. = 381

ALPHA = 1.0

MACH NO. =.6975

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 34

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0404	0.0	1.0706
0.7	-0.7736	1.0	0.5833
1.0	-1.1098	1.8	0.3148
1.6	-1.3570	5.3	-0.0978
2.0	-1.3516	10.2	-0.3130
2.6	-1.1933	15.3	-0.3593
3.1	-1.2538	20.2	-0.4036
3.6	-1.0208	27.7	-0.4461
5.2	-0.6980	37.6	-0.4195
7.7	-0.5763	45.1	-0.3009
10.2	-0.4814	52.6	-0.1345
15.2	-0.4206	60.1	0.0142
20.2	-0.3741	67.5	0.1325
25.2	-0.3487	71.7	0.1869
30.2	-0.3416	85.1	0.2857
40.2	-0.3416	90.1	0.2907
45.2	-0.3770	95.1	0.2470
50.2	-0.3593	100.0	0.0652
52.7	-0.3484		
55.2	-0.3537		
57.7	-0.3432		
60.2	-0.3659		
62.7	-0.3642		
65.2	-0.3917		
67.7	-0.4031		
70.2	-0.4010		
75.2	-0.3709		
80.2	-0.3129		
85.2	-0.2356		
90.1	-0.1425		
95.2	-0.0295		
100.0	0.0836		

	UPPER	LOWER	TOTAL
CN	0.3610	-0.0737	0.2873
CC	-0.0127	0.0131	0.0004
CM	-0.1415	-0.0170	-0.1585

AIRFOIL PERFORMANCE

CL	CD	CM
0.2873	0.0054	-0.1585

Table 5.37.

NPL SECTION ANALYSIS
9510

RUN NO. = 380

ALPHA = 0.0

MACH NO. =.704

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 28

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0708	0.0	1.0879
0.7	-0.4318	1.0	0.4089
1.0	-0.7383	1.8	0.1351
1.6	-0.9139	5.3	-0.2720
2.0	-0.8875	10.2	-0.4703
2.6	-0.7575	15.3	-0.4933
3.1	-0.7258	20.2	-0.5283
3.6	-0.6432	27.7	-0.5510
5.2	-0.4556	37.6	-0.4968
7.7	-0.3955	45.1	-0.3551
10.2	-0.3355	52.6	-0.1795
15.2	-0.3019	60.1	-0.0194
20.2	-0.2754	67.5	0.1040
25.2	-0.2641	71.7	0.1585
30.2	-0.2659	85.1	0.2639
40.2	-0.2851	90.1	0.2748
45.2	-0.3289	95.1	0.2371
50.2	-0.3201	100.0	0.0755
52.7	-0.3049		
55.2	-0.3118		
57.7	-0.3066		
60.2	-0.3325		
62.7	-0.3325		
65.2	-0.3650		
67.7	-0.3793		
70.2	-0.3814		
75.2	-0.3559		
80.2	-0.3057		
85.2	-0.2307		
90.1	-0.1394		
95.2	-0.0227		
100.0	0.0941		

	UPPER	LOWER	TOTAL
CN	0.2915	-0.1416	0.1499
CC	-0.0025	0.0070	0.0046
CM	-0.1257	0.0028	-0.1230

AIRFOIL PERFORMANCE

CL	CD	CM
0.1499	0.0046	-0.1230

Table 5.38.

NPL SECTION ANALYSIS
9510

RUN NO. = 390

ALPHA = 0.0

MACH NO. =.7039

WING DATA FILE NAME = *NPL15.DAT
INPUT FILE NO. - 25

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0755	0.0	1.0880
0.7	-0.4516	1.0	0.4023
1.0	-0.7276	1.8	0.1277
1.6	-0.9075	5.3	-0.2869
2.0	-0.8528	10.2	-0.4863
2.6	-0.7045	15.3	-0.5151
3.1	-0.6868	20.2	-0.5449
3.6	-0.5809	27.7	-0.5571
5.2	-0.4421	37.6	-0.5028
7.7	-0.3890	45.1	-0.3609
10.2	-0.3289	52.6	-0.1767
15.2	-0.2936	60.1	-0.0175
20.2	-0.2670	67.5	0.1063
25.2	-0.2536	71.7	0.1630
30.2	-0.2554	85.1	0.2693
40.2	-0.2729	90.1	0.2767
45.2	-0.3149	95.1	0.2413
50.2	-0.3061	100.0	0.0778
52.7	-0.2894		
55.2	-0.2980		
57.7	-0.2980		
60.2	-0.3221		
62.7	-0.3273		
65.2	-0.3562		
67.7	-0.3715		
70.2	-0.3743		
75.2	-0.3517		
80.2	-0.3021		
85.2	-0.2293		
90.1	-0.1401		
95.2	-0.0223		
100.0	0.0955		

	UPPER	LOWER	TOTAL
CN	0.2827	-0.1451	0.1376
CC	-0.0021	0.0063	0.0041
CM	-0.1226	0.0025	-0.1201

AIRFOIL PERFORMANCE

CL	CD	CM
0.1376	0.0041	-0.1201

Table 5.39.

NPL SECTION ANALYSIS
9510

RUN NO. = 374

ALPHA = 5.0

MACH NO. =.6018

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 7

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.6506	0.0	0.7424
0.7	-2.2141	1.0	0.9708
1.0	-2.1347	1.8	0.7797
1.6	-2.1591	5.3	0.3998
2.0	-2.0559	10.2	0.1516
2.6	-1.8494	15.3	0.0395
3.1	-1.8362	20.2	-0.0439
3.6	-1.8165	27.7	-0.1340
5.2	-1.7103	37.6	-0.1669
7.7	-1.5402	45.1	-0.1208
10.2	-1.3104	52.6	-0.0153
15.2	-1.1159	60.1	0.1074
20.2	-0.9126	67.5	0.1928
25.2	-0.7578	71.7	0.2408
30.2	-0.6523	85.1	0.3145
40.2	-0.5074	90.1	0.2997
45.2	-0.4986	95.1	0.2314
50.2	-0.4503	100.0	-0.0266
52.7	-0.4227		
55.2	-0.4075		
57.7	-0.3857		
60.2	-0.3879		
62.7	-0.3726		
65.2	-0.3754		
67.7	-0.3696		
70.2	-0.3527		
75.2	-0.3127		
80.2	-0.2603		
85.2	-0.2026		
90.1	-0.1386		
95.2	-0.0759		
100.0	-0.0132		

	UPPER	LOWER	TOTAL
CN	0.5988	0.1114	0.7102
CC	-0.0521	0.0266	-0.0255
CM	-0.1810	-0.0650	-0.2460

AIRFOIL PERFORMANCE

CL	CD	CM
0.7097	0.0365	-0.2460

Table 5.40.

NPL SECTION ANALYSIS
9510

RUN NO. = 375

ALPHA = 4.0

MACH NO. =.5981

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 10

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.7328	0.0	0.8067
0.7	-2.0404	1.0	0.9198
1.0	-1.9966	1.8	0.7092
1.6	-2.1496	5.3	0.3191
2.0	-1.9712	10.2	0.0798
2.6	-1.8931	15.3	-0.0290
3.1	-1.7727	20.2	-0.0981
3.6	-1.7304	27.7	-0.1739
5.2	-1.5945	37.6	-0.1962
7.7	-1.3747	45.1	-0.1382
10.2	-1.1215	52.6	-0.0269
15.2	-0.8794	60.1	0.0896
20.2	-0.6862	67.5	0.1849
25.2	-0.5851	71.7	0.2327
30.2	-0.5208	85.1	0.3094
40.2	-0.4433	90.1	0.3043
45.2	-0.4521	95.1	0.2445
50.2	-0.4189	100.0	0.0154
52.7	-0.4009		
55.2	-0.3965		
57.7	-0.3833		
60.2	-0.3943		
62.7	-0.3855		
65.2	-0.3890		
67.7	-0.3891		
70.2	-0.3774		
75.2	-0.3353		
80.2	-0.2788		
85.2	-0.2125		
90.1	-0.1363		
95.2	-0.0530		
100.0	0.0303		

	UPPER	LOWER	TOTAL
CN	0.5337	0.0867	0.6204
CC	-0.0452	0.0246	-0.0206
CM	-0.1675	-0.0600	-0.2275

AIRFOIL PERFORMANCE

CL	CD	CM
0.6203	0.0228	-0.2275

Table 5.41.

NPL SECTION ANALYSIS
9510

RUN NO. = 376

ALPHA = 3.0

MACH NO. =.6048

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 15

UPPER SURFACE		LOWER SURFACE	
%CHORD	CP LOCAL	%CHORD	CP LOCAL
0.0	0.8222	0.0	0.8858
0.7	-1.7883	1.0	0.8510
1.0	-2.1089	1.8	0.6203
1.6	-2.0838	5.3	0.2220
2.0	-1.9407	10.2	-0.0087
2.6	-1.8966	15.3	-0.1027
3.1	-1.7116	20.2	-0.1639
3.6	-1.5618	27.7	-0.2338
5.2	-1.3761	37.6	-0.2447
7.7	-1.1275	45.1	-0.1770
10.2	-0.8680	52.6	-0.0501
15.2	-0.6717	60.1	0.0696
20.2	-0.5496	67.5	0.1671
25.2	-0.5068	71.7	0.2183
30.2	-0.4695	85.1	0.3014
40.2	-0.4212	90.1	0.2976
45.2	-0.4388	95.1	0.2448
50.2	-0.4147	100.0	0.0352
52.7	-0.3908		
55.2	-0.3930		
57.7	-0.3822		
60.2	-0.4016		
62.7	-0.3930		
65.2	-0.3979		
67.7	-0.4047		
70.2	-0.3958		
75.2	-0.3559		
80.2	-0.2991		
85.2	-0.2254		
90.1	-0.1438		
95.2	-0.0461		
100.0	0.0515		

	UPPER	LOWER	TOTAL
CN	0.4860	0.0499	0.5358
CC	-0.0378	0.0224	-0.0153
CM	-0.1620	-0.0496	-0.2116

AIRFOIL PERFORMANCE

CL	CD	CM
0.5359	0.0127	-0.2116

Table 5.42.

NPL SECTION ANALYSIS
9510

RUN NO. = 377

ALPHA = 2.0

MACH NO. =.5985

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 18

UPPER SURFACE		LOWER SURFACE	
%CHORD	CP LOCAL	%CHORD	CP LOCAL
0.0	0.9240	0.0	0.9689
0.7	-1.4059	1.0	0.7211
1.0	-1.8057	1.8	0.4690
1.6	-1.7258	5.3	0.0686
2.0	-1.6058	10.2	-0.1438
2.6	-1.5059	15.3	-0.2141
3.1	-1.4015	20.2	-0.2649
3.6	-1.1127	27.7	-0.3069
5.2	-0.9281	37.6	-0.3046
7.7	-0.7367	45.1	-0.2208
10.2	-0.5920	52.6	-0.0957
15.2	-0.5008	60.1	0.0356
20.2	-0.4362	67.5	0.1447
25.2	-0.4034	71.7	0.1947
30.2	-0.3812	85.1	0.2849
40.2	-0.3613	90.1	0.2872
45.2	-0.3812	95.1	0.2377
50.2	-0.3635	100.0	0.0531
52.7	-0.3576		
55.2	-0.3620		
57.7	-0.3488		
60.2	-0.3687		
62.7	-0.3665		
65.2	-0.3772		
67.7	-0.3873		
70.2	-0.3811		
75.2	-0.3506		
80.2	-0.2977		
85.2	-0.2286		
90.1	-0.1444		
95.2	-0.0365		
100.0	0.0714		

	UPPER	LOWER	TOTAL
CN	0.3995	-0.0053	0.3942
CC	-0.0251	0.0182	-0.0069
CM	-0.1450	-0.0337	-0.1786

AIRFOIL PERFORMANCE

CL	CD	CM
0.3942	0.0068	-0.1786

Table 5.43.

NPL SECTION ANALYSIS
9510

RUN NO. = 378

ALPHA = 1.0

MACH NO. =.6

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 22

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9966	0.0	1.0268
0.7	-0.9988	1.0	0.5776
1.0	-1.3215	1.8	0.3120
1.6	-1.3357	5.3	-0.0841
2.0	-1.2277	10.2	-0.2700
2.6	-1.0954	15.3	-0.3193
3.1	-1.0183	20.2	-0.3526
3.6	-0.7957	27.7	-0.3814
5.2	-0.6608	37.6	-0.3681
7.7	-0.5522	45.1	-0.2705
10.2	-0.4546	52.6	-0.1267
15.2	-0.3947	60.1	0.0089
20.2	-0.3482	67.5	0.1256
25.2	-0.3284	71.7	0.1794
30.2	-0.3196	85.1	0.2727
40.2	-0.3152	90.1	0.2768
45.2	-0.3394	95.1	0.2355
50.2	-0.3284	100.0	0.0620
52.7	-0.3249		
55.2	-0.3293		
57.7	-0.3227		
60.2	-0.3425		
62.7	-0.3425		
65.2	-0.3570		
67.7	-0.3656		
70.2	-0.3647		
75.2	-0.3397		
80.2	-0.2933		
85.2	-0.2256		
90.1	-0.1444		
95.2	-0.0323		
100.0	0.0799		

	UPPER	LOWER	TOTAL
CN	0.3371	-0.0571	0.2799
CC	-0.0143	0.0133	-0.0010
CM	-0.1319	-0.0195	-0.1515

AIRFOIL PERFORMANCE

CL	CD	CM
0.2799	0.0039	-0.1515

Table 5.44.

NPL SECTION ANALYSIS
9510

RUN NO. = 379

ALPHA = 0.0

MACH NO. =.5948

WING DATA FILE NAME = *NPL14.DAT
INPUT FILE NO. - 28

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0362	0.0	1.0513
0.7	-0.5491	1.0	0.3758
1.0	-0.8104	1.8	0.1096
1.6	-0.8817	5.3	-0.2595
2.0	-0.8058	10.2	-0.4093
2.6	-0.6875	15.3	-0.4281
3.1	-0.6474	20.2	-0.4505
3.6	-0.5715	27.7	-0.4528
5.2	-0.4304	37.6	-0.4147
7.7	-0.3766	45.1	-0.3048
10.2	-0.3205	52.6	-0.1611
15.2	-0.2847	60.1	-0.0159
20.2	-0.2555	67.5	0.0988
25.2	-0.2455	71.7	0.1532
30.2	-0.2455	85.1	0.2512
40.2	-0.2589	90.1	0.2607
45.2	-0.2857	95.1	0.2231
50.2	-0.2813	100.0	0.0707
52.7	-0.2823		
55.2	-0.2890		
57.7	-0.2823		
60.2	-0.3001		
62.7	-0.3001		
65.2	-0.3237		
67.7	-0.3368		
70.2	-0.3368		
75.2	-0.3188		
80.2	-0.2781		
85.2	-0.2176		
90.1	-0.1400		
95.2	-0.0264		
100.0	0.0872		

	UPPER	LOWER	TOTAL
CN	0.2683	-0.1133	0.1550
CC	-0.0035	0.0065	0.0030
CM	-0.1148	-0.0037	-0.1185

AIRFOIL PERFORMANCE

CL	CD	CM
0.1550	0.0030	-0.1185

Table 5.45.

NPL SECTION ANALYSIS
9510

RUN NO. = 371

ALPHA = 5.0

MACH NO. =.5079

WING DATA FILE NAME = *NPL13.DAT
INPUT FILE NO. - 41

UPPER SURFACE %CHORD	CP LOCAL	LOWER SURFACE %CHORD	CP LOCAL
0.0	0.4722	0.0	0.6109
0.7	-2.3670	1.0	0.9768
1.0	-2.2308	1.8	0.8011
1.6	-2.3502	5.3	0.4265
2.0	-2.0887	10.2	0.1787
2.6	-1.8796	15.3	0.0634
3.1	-1.8244	20.2	-0.0144
3.6	-1.8592	27.7	-0.1037
5.2	-1.7343	37.6	-0.1354
7.7	-1.5281	45.1	-0.0951
10.2	-1.2521	52.6	-0.0086
15.2	-0.9906	60.1	0.1204
20.2	-0.7786	67.5	0.1993
25.2	-0.6391	71.7	0.2436
30.2	-0.5646	85.1	0.3112
40.2	-0.4614	90.1	0.3003
45.2	-0.4586	95.1	0.2377
50.2	-0.4270	100.0	-0.0058
52.7	-0.4167		
55.2	-0.4052		
57.7	-0.3822		
60.2	-0.3908		
62.7	-0.3765		
65.2	-0.3678		
67.7	-0.3617		
70.2	-0.3478		
75.2	-0.3084		
80.2	-0.2574		
85.2	-0.1982		
90.1	-0.1356		
95.2	-0.0655		
100.0	0.0046		

	UPPER	LOWER	TOTAL
CN	0.5662	0.1267	0.6929
CC	-0.0541	0.0259	-0.0282
CM	-0.1703	-0.0702	-0.2405

AIRFOIL PERFORMANCE

CL	CD	CM
0.6927	0.0323	-0.2405

Table 5.46.

NPL SECTION ANALYSIS
9510

RUN NO. = 373

ALPHA = 4.9

MACH NO. =.5

WING DATA FILE NAME = *NPL13.DAT
INPUT FILE NO. - 47

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.4779	0.0	0.6119
0.7	-2.3543	1.0	0.9697
1.0	-2.2458	1.8	0.7893
1.6	-2.3544	5.3	0.4109
2.0	-2.1285	10.2	0.1626
2.6	-1.8580	15.3	0.0503
3.1	-1.7926	20.2	-0.0266
3.6	-1.8669	27.7	-0.1094
5.2	-1.7609	37.6	-0.1449
7.7	-1.5326	45.1	-0.1005
10.2	-1.2273	52.6	-0.0207
15.2	-0.9368	60.1	0.1123
20.2	-0.7382	67.5	0.1937
25.2	-0.6250	71.7	0.2416
30.2	-0.5572	85.1	0.3073
40.2	-0.4658	90.1	0.2978
45.2	-0.4628	95.1	0.2356
50.2	-0.4334	100.0	-0.0036
52.7	-0.4134		
55.2	-0.4046		
57.7	-0.3870		
60.2	-0.3929		
62.7	-0.3782		
65.2	-0.3789		
67.7	-0.3745		
70.2	-0.3613		
75.2	-0.3208		
80.2	-0.2671		
85.2	-0.2075		
90.1	-0.1395		
95.2	-0.0662		
100.0	0.0071		

	UPPER	LOWER	TOTAL
CN	0.5637	0.1192	0.6829
CC	-0.0534	0.0257	-0.0277
CM	-0.1716	-0.0676	-0.2392

AIRFOIL PERFORMANCE

CL	CD	CM
0.6828	0.0308	-0.2392

Table 5.47

NPL SECTION ANALYSIS
9510

RUN NO. = 370

ALPHA = 4.0

MACH NO. =.5009

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

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.5795	0.0	0.6688
0.7	-2.6020	1.0	0.9364
1.0	-2.3782	1.8	0.7386
1.6	-2.6987	5.3	0.3490
2.0	-2.4834	10.2	0.1076
2.6	-1.9831	15.3	0.0000
3.1	-1.9709	20.2	-0.0729
3.6	-1.8224	27.7	-0.1545
5.2	-1.6367	37.6	-0.1779
7.7	-1.2809	45.1	-0.1341
10.2	-1.0272	52.6	-0.0490
15.2	-0.8354	60.1	0.1010
20.2	-0.7116	67.5	0.1805
25.2	-0.5379	71.7	0.2276
30.2	-0.4974	85.1	0.3043
40.2	-0.4454	90.1	0.2952
45.2	-0.4512	95.1	0.2368
50.2	-0.4280	100.0	0.0163
52.7	-0.4102		
55.2	-0.4045		
57.7	-0.3844		
60.2	-0.3959		
62.7	-0.3815		
65.2	-0.4155		
67.7	-0.3908		
70.2	-0.3802		
75.2	-0.3428		
80.2	-0.2878		
85.2	-0.2211		
90.1	-0.1451		
95.2	-0.0592		
100.0	0.0267		

	UPPER	LOWER	TOTAL
CN	0.5402	0.0932	0.6335
CC	-0.0534	0.0249	-0.0286
CM	-0.1689	-0.0600	-0.2289

AIRFOIL PERFORMANCE

CL	CD	CM
0.6339	0.0157	-0.2289

Table 5.48.

NPL SECTION ANALYSIS
9510

RUN NO. = 369

ALPHA = 3.0

MACH NO. =.4933

WING DATA FILE NAME = *NPL13.DAT
INPUT FILE NO. - 35

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.7283	0.0	0.8066
0.7	-2.0615	1.0	0.8401
1.0	-2.1879	1.8	0.6118
1.6	-2.0130	5.3	0.2161
2.0	-1.8953	10.2	-0.0061
2.6	-1.6207	15.3	-0.0913
3.1	-1.4516	20.2	-0.1522
3.6	-1.3641	27.7	-0.2100
5.2	-1.1564	37.6	-0.2192
7.7	-0.9075	45.1	-0.1613
10.2	-0.7102	52.6	-0.0519
15.2	-0.5858	60.1	0.0641
20.2	-0.5069	67.5	0.1678
25.2	-0.4501	71.7	0.2183
30.2	-0.4261	85.1	0.2971
40.2	-0.3901	90.1	0.2959
45.2	-0.3961	95.1	0.2452
50.2	-0.3781	100.0	0.0501
52.7	-0.3803		
55.2	-0.3772		
57.7	-0.3591		
60.2	-0.3682		
62.7	-0.3622		
65.2	-0.3622		
67.7	-0.3656		
70.2	-0.3583		
75.2	-0.3268		
80.2	-0.2758		
85.2	-0.2115		
90.1	-0.1338		
95.2	-0.0349		
100.0	0.0640		

	UPPER	LOWER	TOTAL
CN	0.4376	0.0552	0.4928
CC	-0.0378	0.0214	-0.0164
CM	-0.1474	-0.0514	-0.1988

AIRFOIL PERFORMANCE

CL	CD	CM
0.4930	0.0094	-0.1988

Table 5.49.

NPL SECTION ANALYSIS
9510

RUN NO. = 368

ALPHA = 2.0

MACH NO. =.4997

WING DATA FILE NAME = *NPL.DAT
INPUT FILE NO. - 13

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.8663	0.0	0.9021
0.7	-1.6018	1.0	0.7257
1.0	-1.7936	1.8	0.4771
1.6	-1.7569	5.3	0.0752
2.0	-1.6527	10.2	-0.1272
2.6	-1.4932	15.3	-0.2000
3.1	-1.3001	20.2	-0.2464
3.6	-1.0824	27.7	-0.2899
5.2	-0.9341	37.6	-0.2870
7.7	~~-0.7825~~	45.1	~~-0.2203~~
10.2	-0.6526	52.6	-0.0814
15.2	-0.5722	60.1	0.0349
20.2	-0.5134	67.5	0.1313
25.2	-0.3895	71.7	0.1829
30.2	-0.3750	85.1	0.2709
40.2	-0.3547	90.1	0.2705
45.2	-0.3721	95.1	0.2234
50.2	-0.3634	100.0	0.0423
52.7	-0.3547		
55.2	-0.3547		
57.7	-0.3460		
60.2	-0.3604		
62.7	-0.3575		
65.2	-0.3916		
67.7	-0.3787		
70.2	-0.3728		
75.2	-0.3445		
80.2	-0.2975		
85.2	-0.2305		
90.1	-0.1529		
95.2	-0.0484		
100.0	0.0560		

	UPPER	LOWER	TOTAL
CN	0.4092	-0.0033	0.4059
CC	-0.0278	0.0182	-0.0097
CM	-0.1465	-0.0316	-0.1781

AIRFOIL PERFORMANCE

CL	CD	CM
0.4060	0.0045	-0.1781

Table 5.50.

NPL SECTION ANALYSIS
9510

RUN NO. = 367

ALPHA = 1.0

MACH NO. =.4977

WING DATA FILE NAME = *NPL13.DAT
INPUT FILE NO. - 30

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	0.9649	0.0	0.9920
0.7	-1.0216	1.0	0.5422
1.0	-1.2517	1.8	0.2800
1.6	-1.2227	5.3	-0.0923
2.0	-1.1326	10.2	-0.2621
2.6	-0.9914	15.3	-0.2962
3.1	-0.8802	20.2	-0.3318
3.6	-0.7390	27.7	-0.3584
5.2	-0.6159	37.6	-0.3407
7.7	-0.5197	45.1	-0.2518
10.2	-0.4386	52.6	-0.1188
15.2	-0.3875	60.1	0.0119
20.2	-0.3425	67.5	0.1150
25.2	-0.3187	71.7	0.1666
30.2	-0.3128	85.1	0.2564
40.2	-0.3068	90.1	0.2596
45.2	-0.3247	95.1	0.2211
50.2	-0.3187	100.0	0.0613
52.7	-0.3217		
55.2	-0.3247		
57.7	-0.3128		
60.2	-0.3307		
62.7	-0.3277		
65.2	-0.3378		
67.7	-0.3434		
70.2	-0.3410		
75.2	-0.3181		
80.2	-0.2747		
85.2	-0.2133		
90.1	-0.1386		
95.2	-0.0319		
100.0	0.0747		

	UPPER	LOWER	TOTAL
CN	0.3226	-0.0541	0.2685
CC	-0.0138	0.0123	-0.0015
CM	-0.1263	-0.0185	-0.1448

AIRFOIL PERFORMANCE

CL	CD	CM
0.2685	0.0032	-0.1448

Table 5.51.

NPL SECTION ANALYSIS
9510

RUN NO. = 366

ALPHA = 0.0

MACH NO. =.4963

WING DATA FILE NAME = *NPL13.DAT
INPUT FILE NO. - 27

UPPER SURFACE %CHORD	UPPER SURFACE CP LOCAL	LOWER SURFACE %CHORD	LOWER SURFACE CP LOCAL
0.0	1.0208	0.0	1.0265
0.7	-0.5380	1.0	0.3062
1.0	-0.7649	1.8	0.0467
1.6	-0.8256	5.3	-0.2887
2.0	-0.7666	10.2	-0.4170
2.6	-0.6398	15.3	-0.4357
3.1	-0.5985	20.2	-0.4532
3.6	-0.5337	27.7	-0.4620
5.2	-0.4162	37.6	-0.4269
7.7	-0.3687	45.1	-0.3217
10.2	-0.3211	52.6	-0.1794
15.2	-0.2973	60.1	-0.0382
20.2	-0.2765	67.5	0.0793
25.2	-0.2573	71.7	0.1301
30.2	-0.2603	85.1	0.2307
40.2	-0.2778	90.1	0.2391
45.2	-0.3012	95.1	0.2058
50.2	-0.3012	100.0	0.0607
52.7	-0.2929		
55.2	-0.3016		
57.7	-0.2958		
60.2	-0.3132		
62.7	-0.3132		
65.2	-0.3271		
67.7	-0.3336		
70.2	-0.3336		
75.2	-0.3135		
80.2	-0.2732		
85.2	-0.2165		
90.1	-0.1431		
95.2	-0.0317		
100.0	0.0798		

	UPPER	LOWER	TOTAL
CN	0.2741	-0.1310	0.1431
CC	-0.0032	0.0056	0.0024
CM	-0.1178	0.0053	-0.1125

AIRFOIL PERFORMANCE

CL	CD	CM
0.1431	0.0024	-0.1125

Table 5.52.

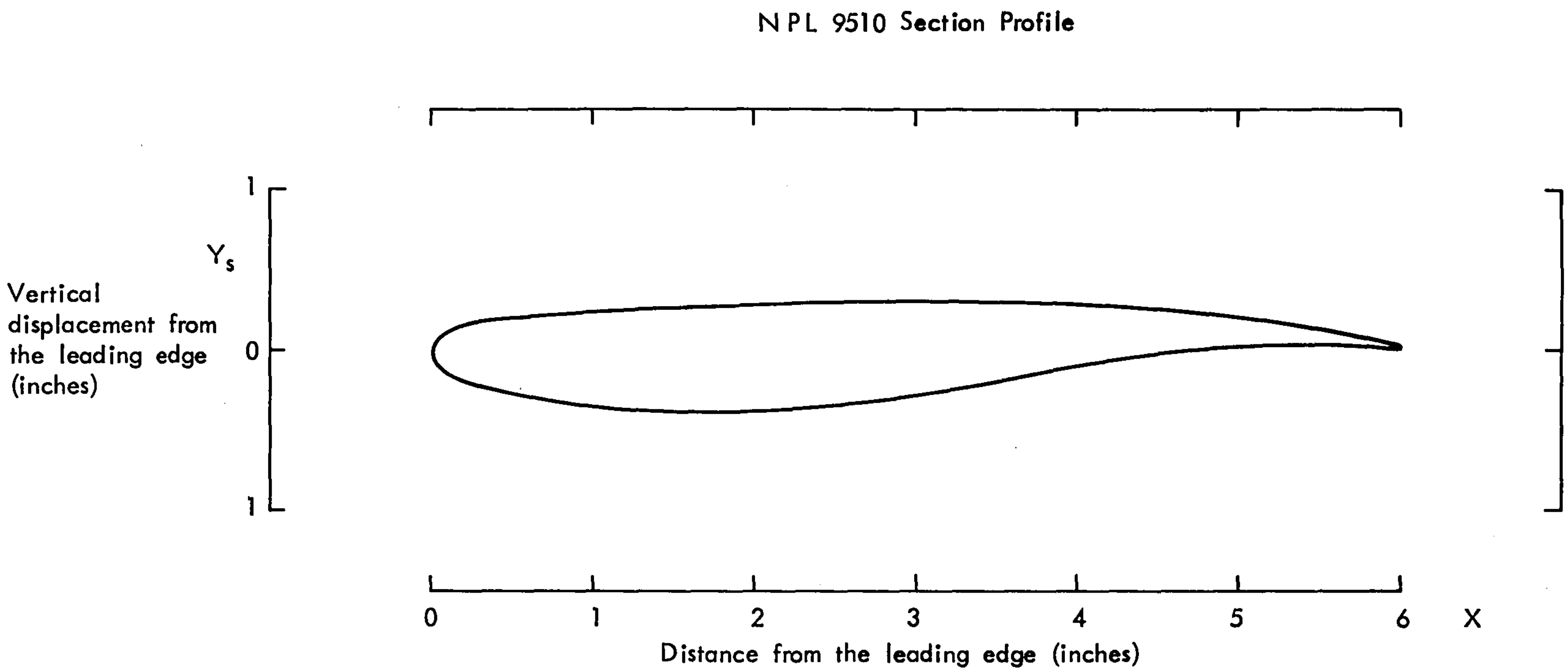
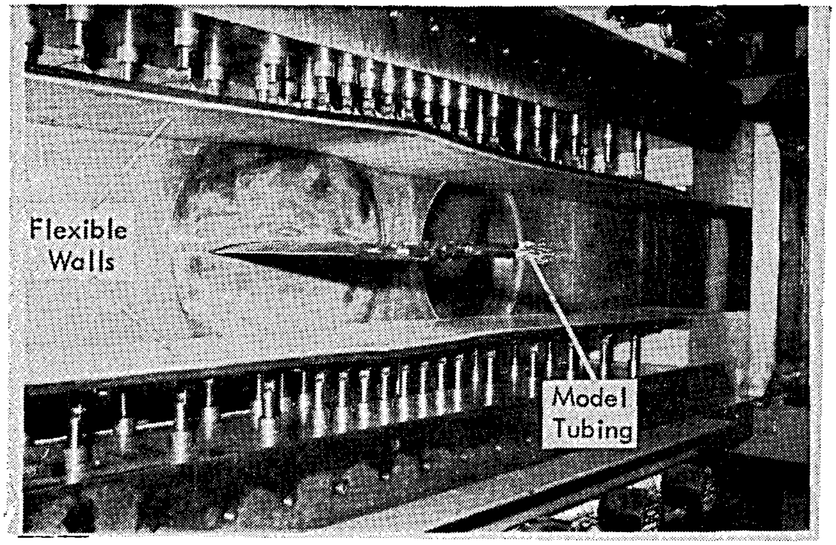


FIG. 1 MODEL PROFILE



Walls streamlined for the test condition $M_\infty \simeq 0.87$; $\alpha = 2^\circ$

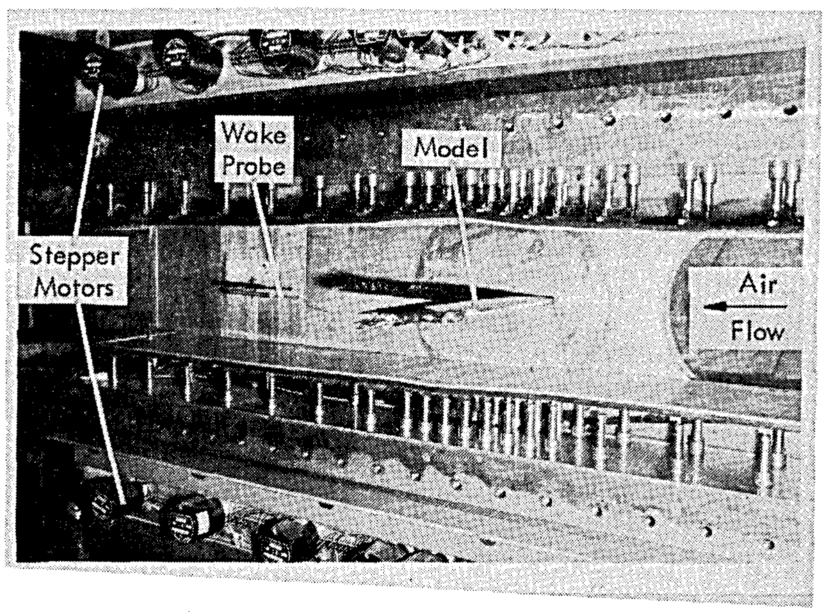


FIG. 2 NPL 9510 MODEL AND WAKE PROBE MOUNTED IN TSWT

Figure 3: NPL 9510 Model Pressure Distributions

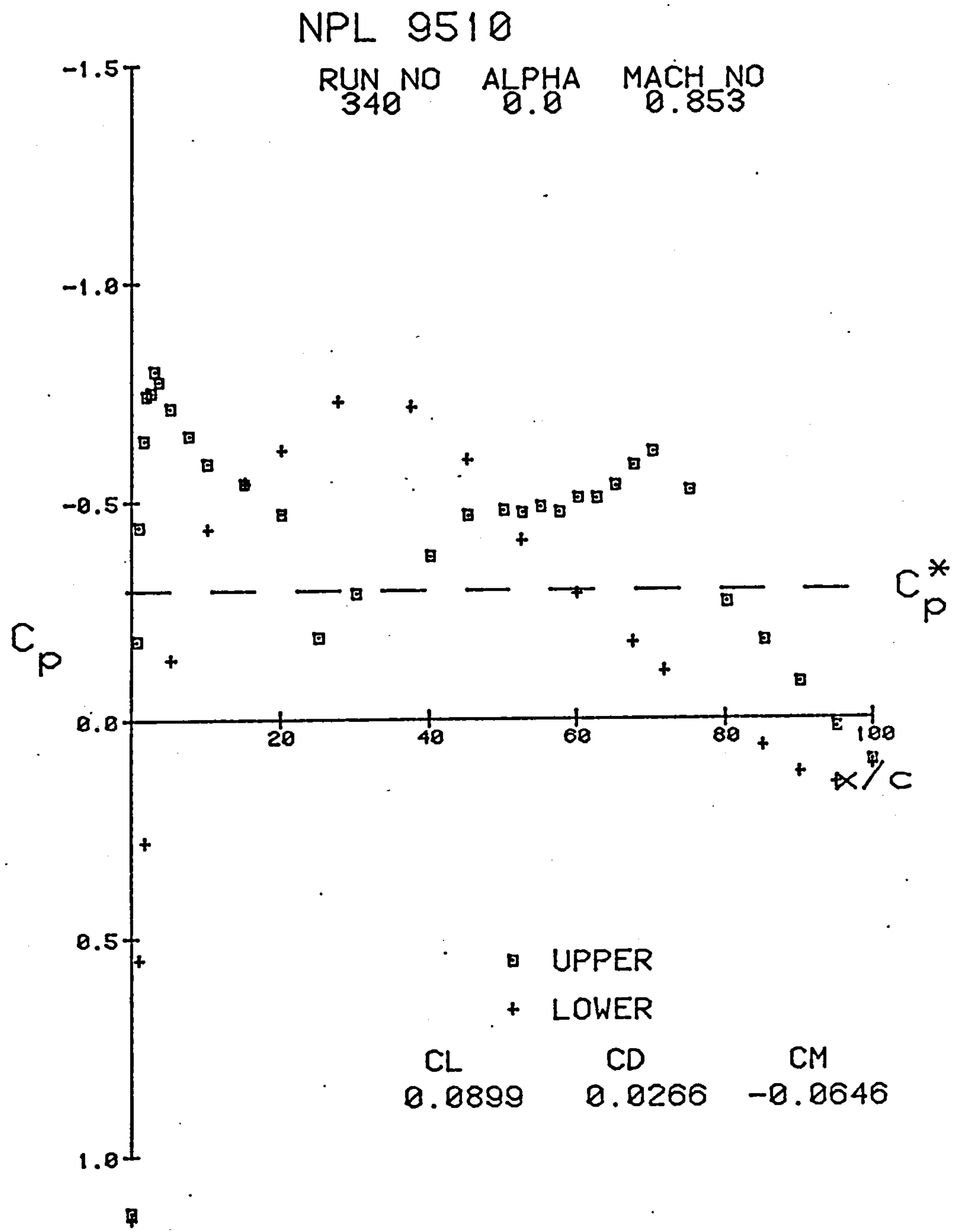
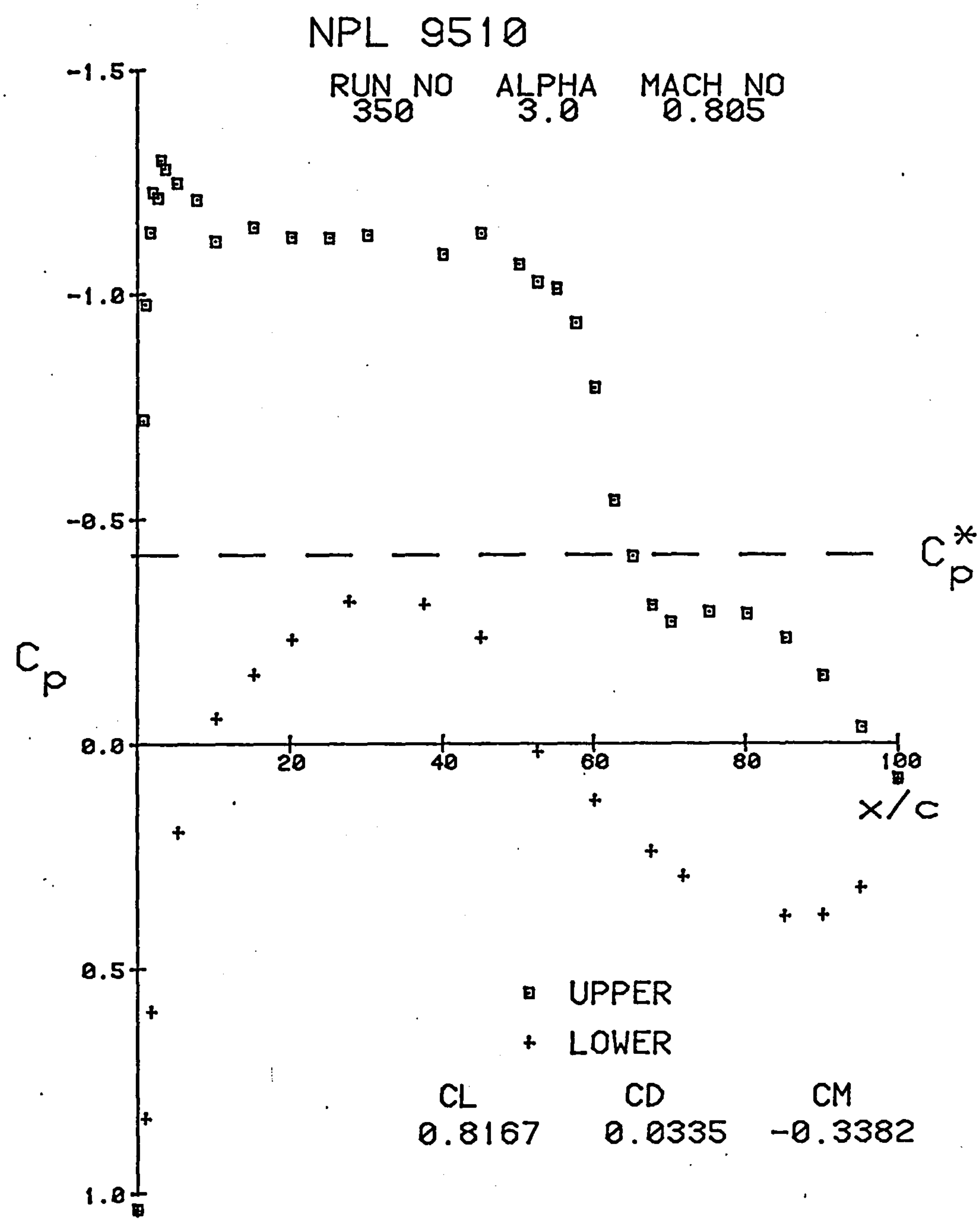

NPL 9510
RUN NO 340
ALPHA 0.0
MACH NO 0.853
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.0899
CD 0.0266
CM -0.0646

Fig. 3.1



NPL 9510
RUN NO 350
ALPHA 3.0
MACH NO 0.805
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.8167
CD 0.0335
CM -0.3382

Fig. 3.2

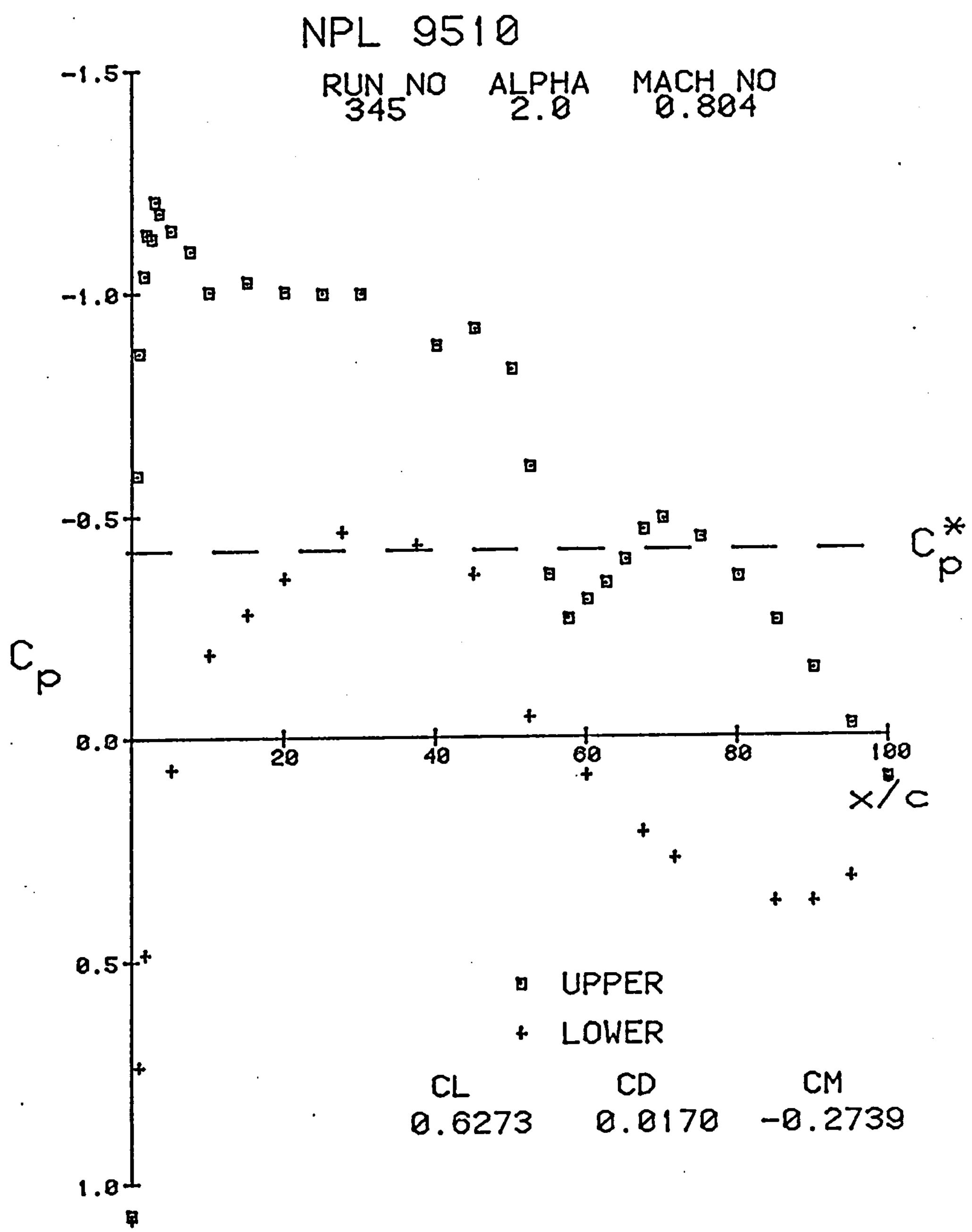

NPL 9510
RUN NO 345
ALPHA 2.0
MACH NO 0.804
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
C_p^*
UPPER
LOWER
CL 0.6273
CD 0.0170
CM -0.2739

Fig. 3.3

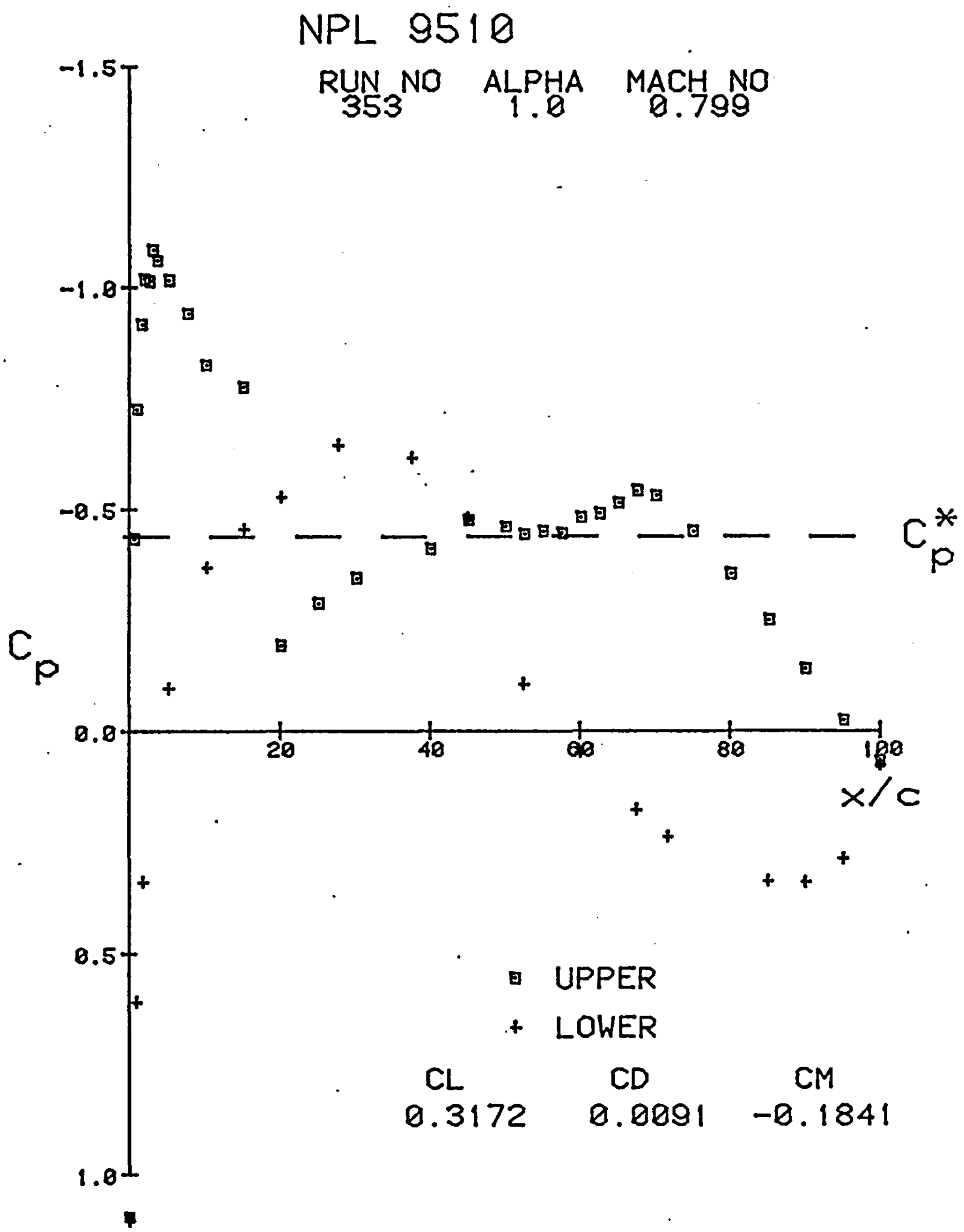

NPL 9510
RUN NO 353
ALPHA 1.0
MACH NO 0.799
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.3172
CD 0.0091
CM -0.1841

Fig. 3.4.

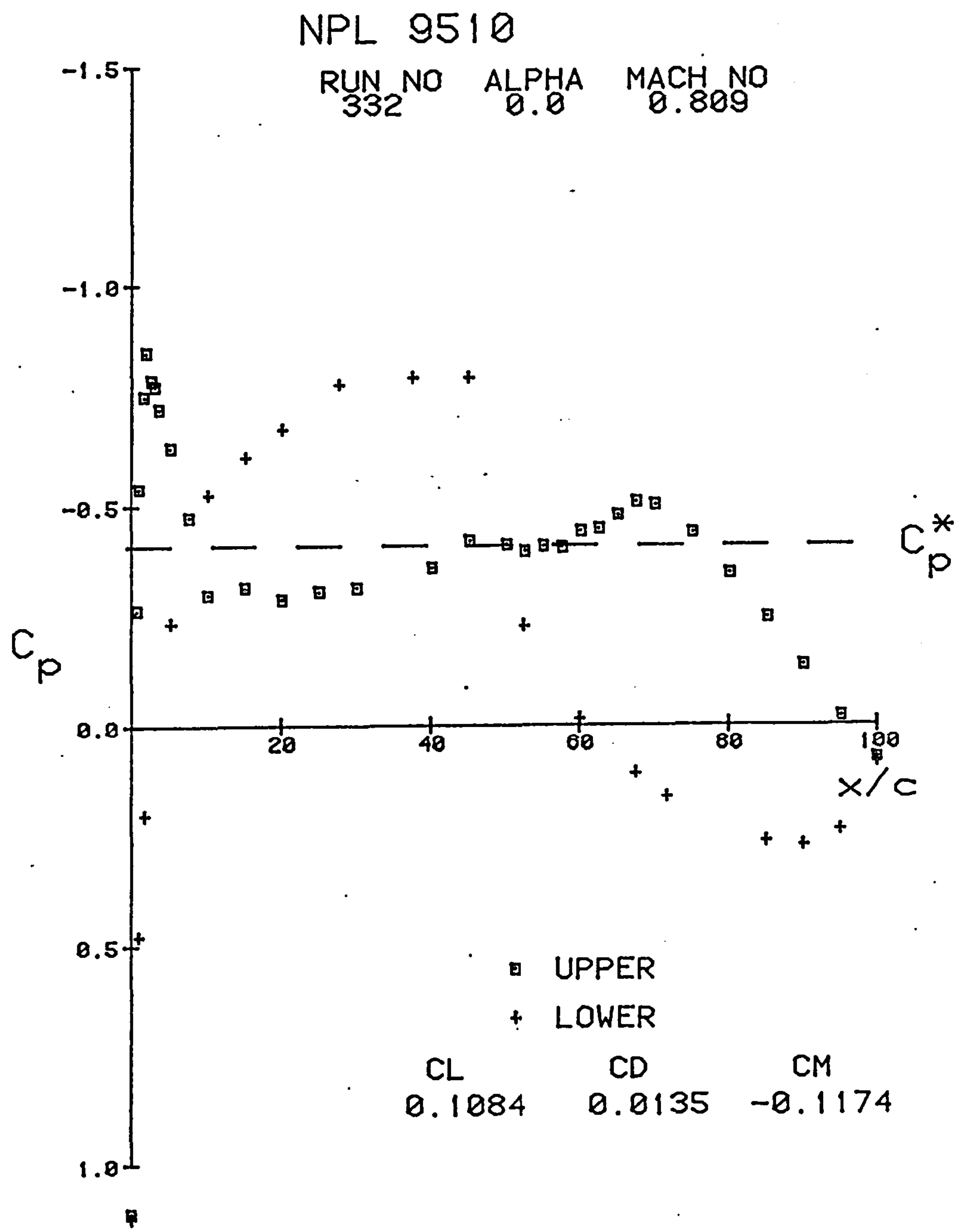

NPL 9510
RUN NO 332
ALPHA 0.0
MACH NO 0.809
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1084
CD 0.0135
CM -0.1174

Fig. 3.5.

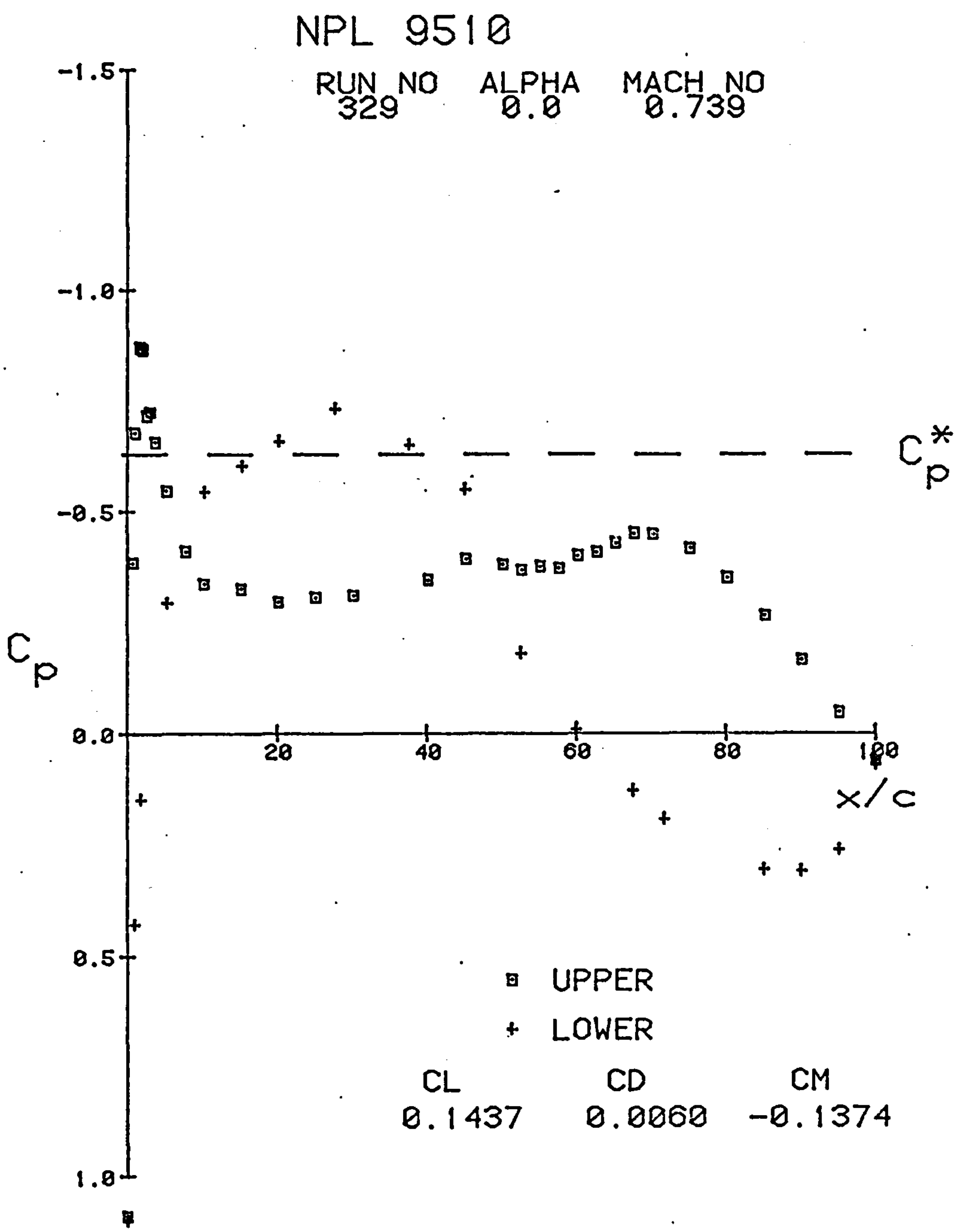

NPL 9510
RUN NO 329
ALPHA 0.0
MACH NO 0.739
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1437
CD 0.0060
CM -0.1374

Fig. 3.6.

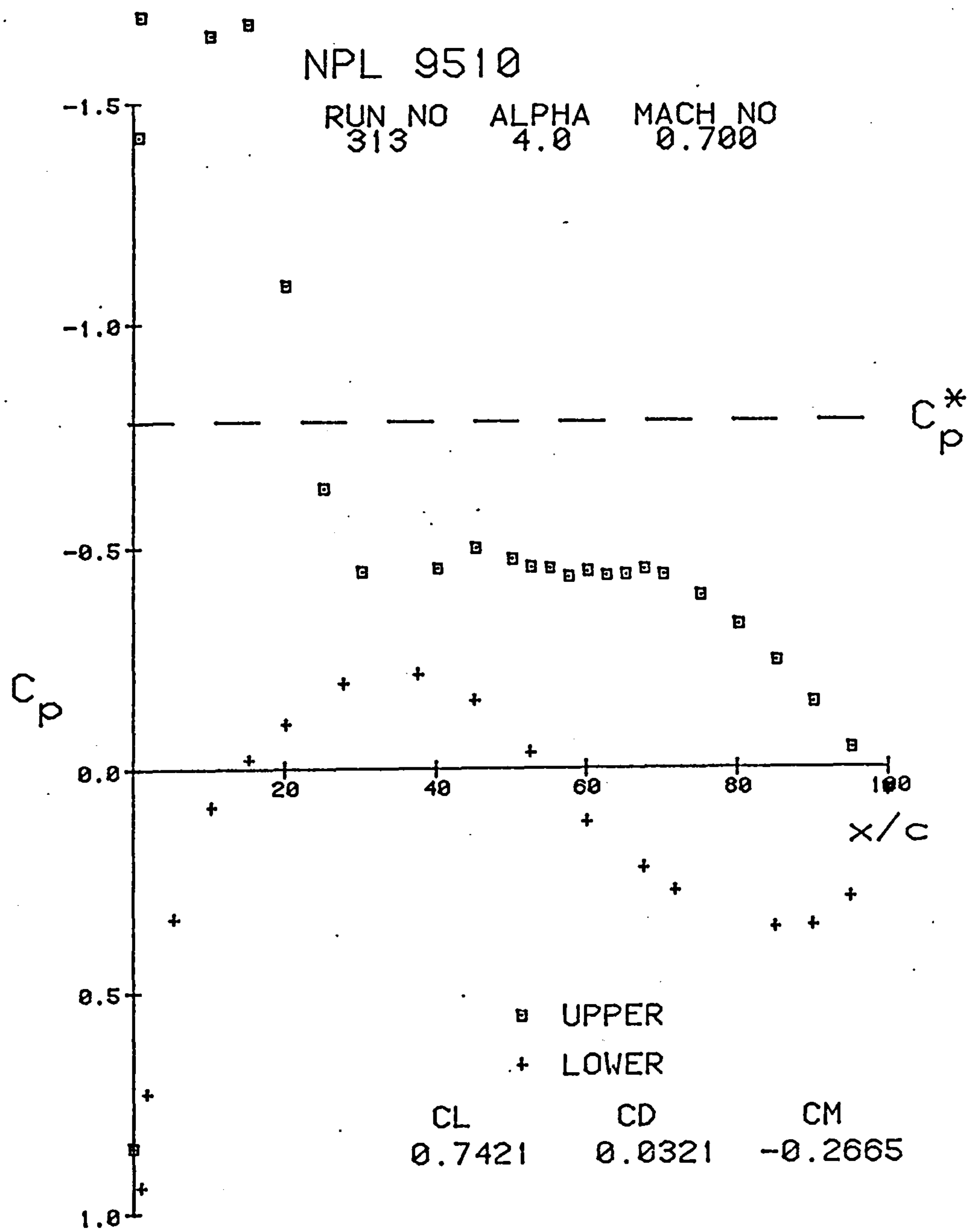

NPL 9510
RUN NO 313
ALPHA 4.0
MACH NO 0.700
-1.5
-1.0
-0.5
0.0
0.5
1.0
C*p
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.7421
CD 0.0321
CM -0.2665

Fig. 3.7.

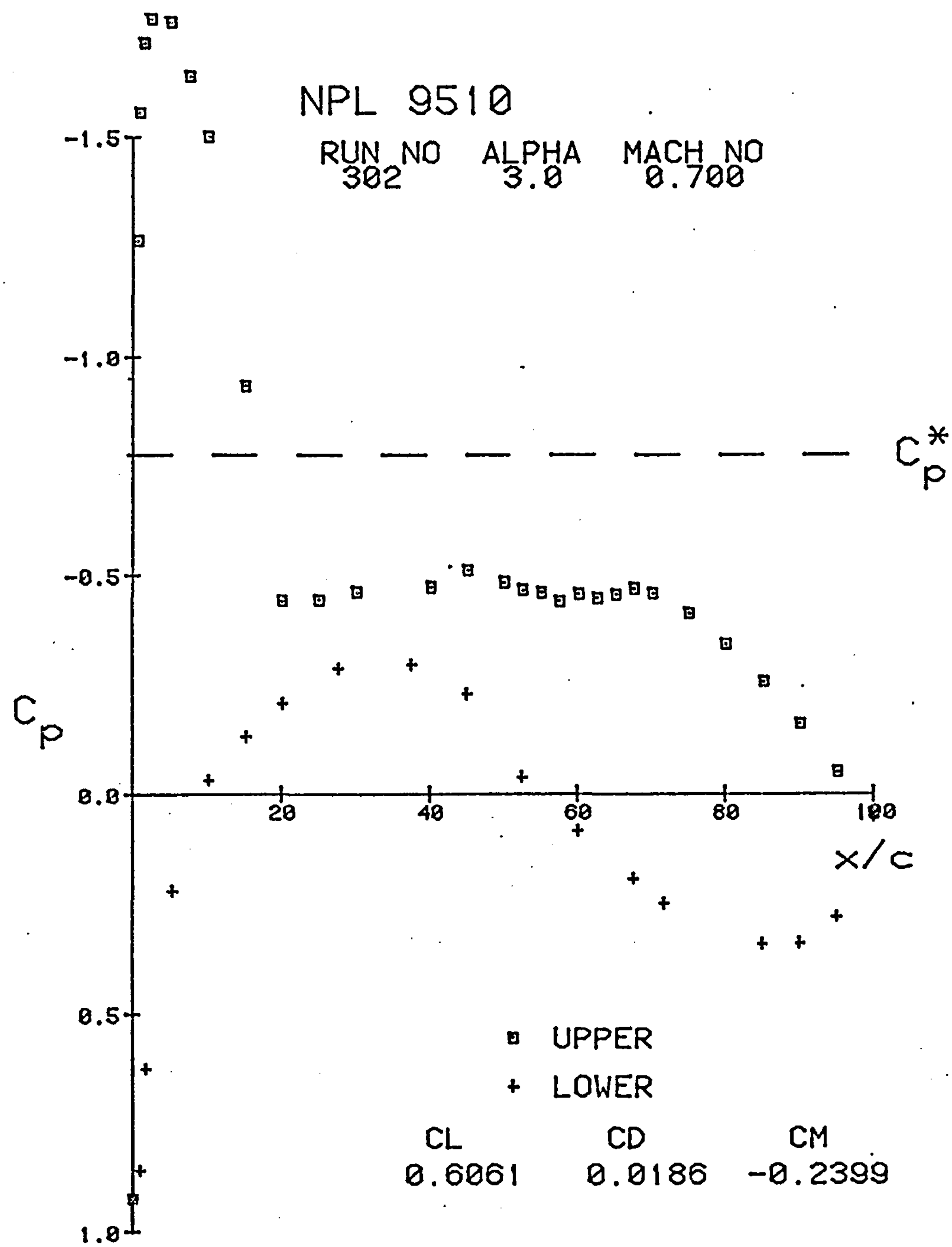

NPL 9510
RUN NO 302
ALPHA 3.0
MACH NO 0.700
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p^*
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6061
CD 0.0186
CM -0.2399

Fig. 3.8.

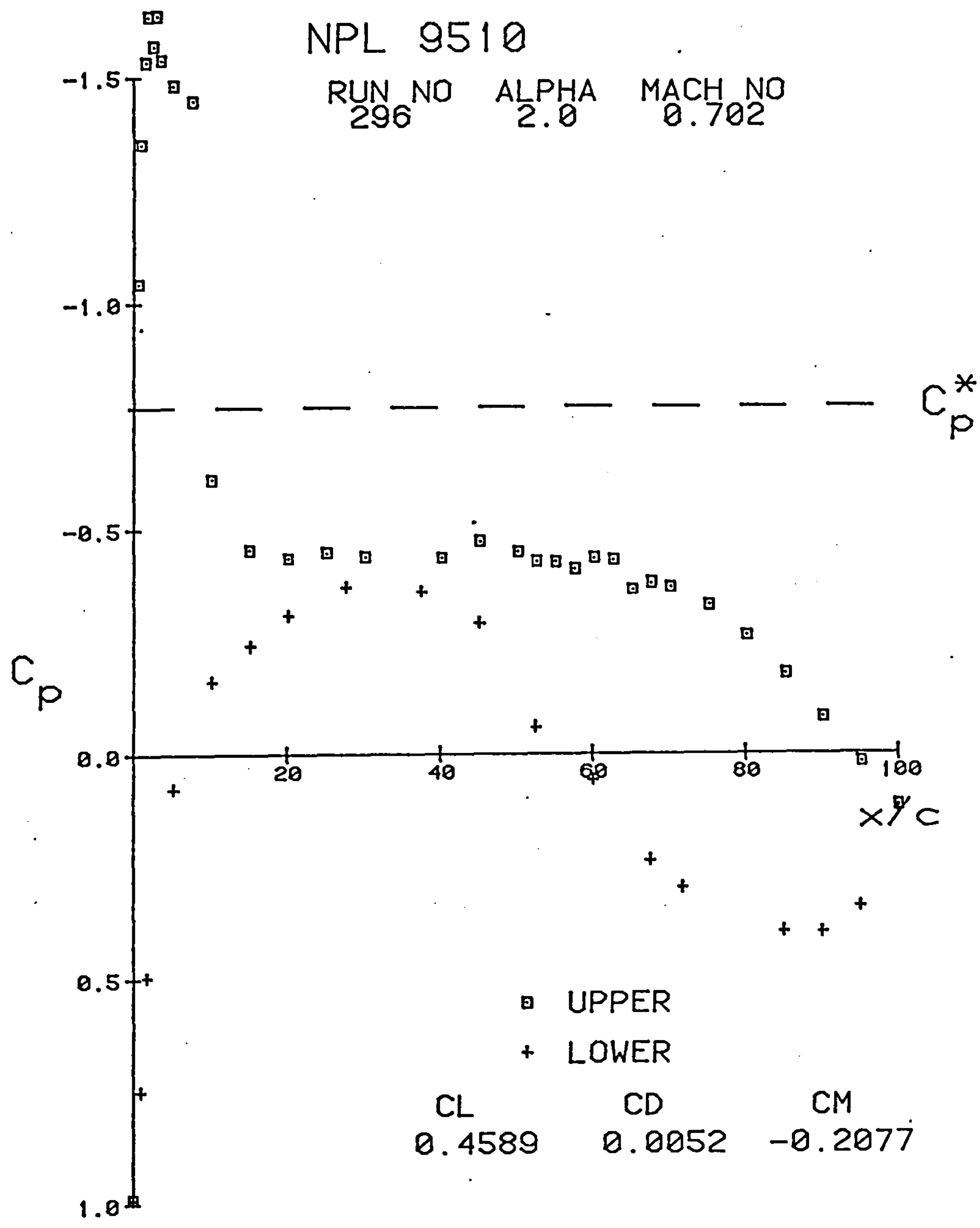

NPL 9510
RUN NO 296
ALPHA 2.0
MACH NO 0.702
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4589
CD 0.0052
CM -0.2077

Fig. 3.9.

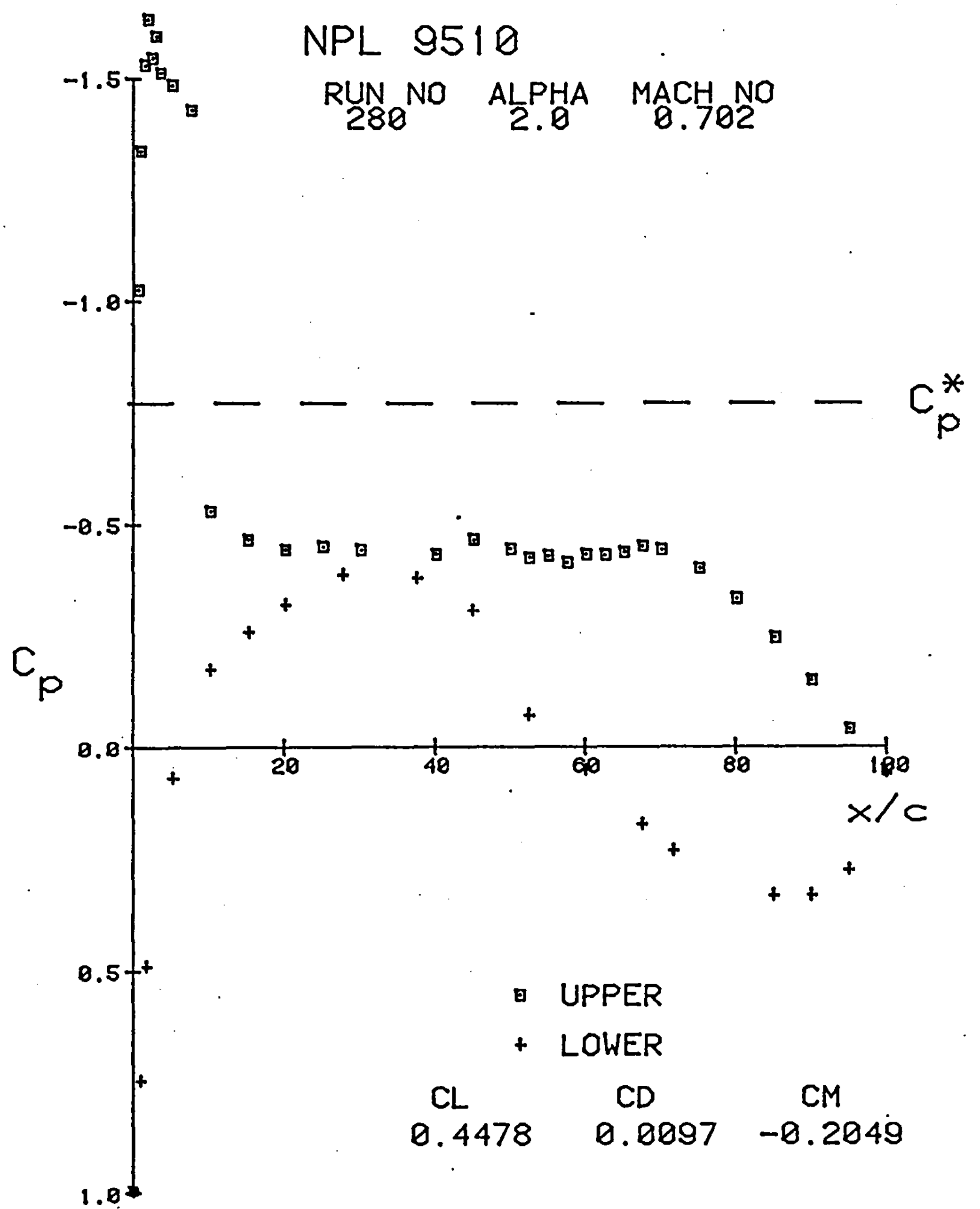

NPL 9510
RUN NO 280
ALPHA 2.0
MACH NO 0.702
-1.5
-1.0
-0.5
0.0
0.5
1.0
C*p
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4478
CD 0.0097
CM -0.2049

Fig. 3.10.

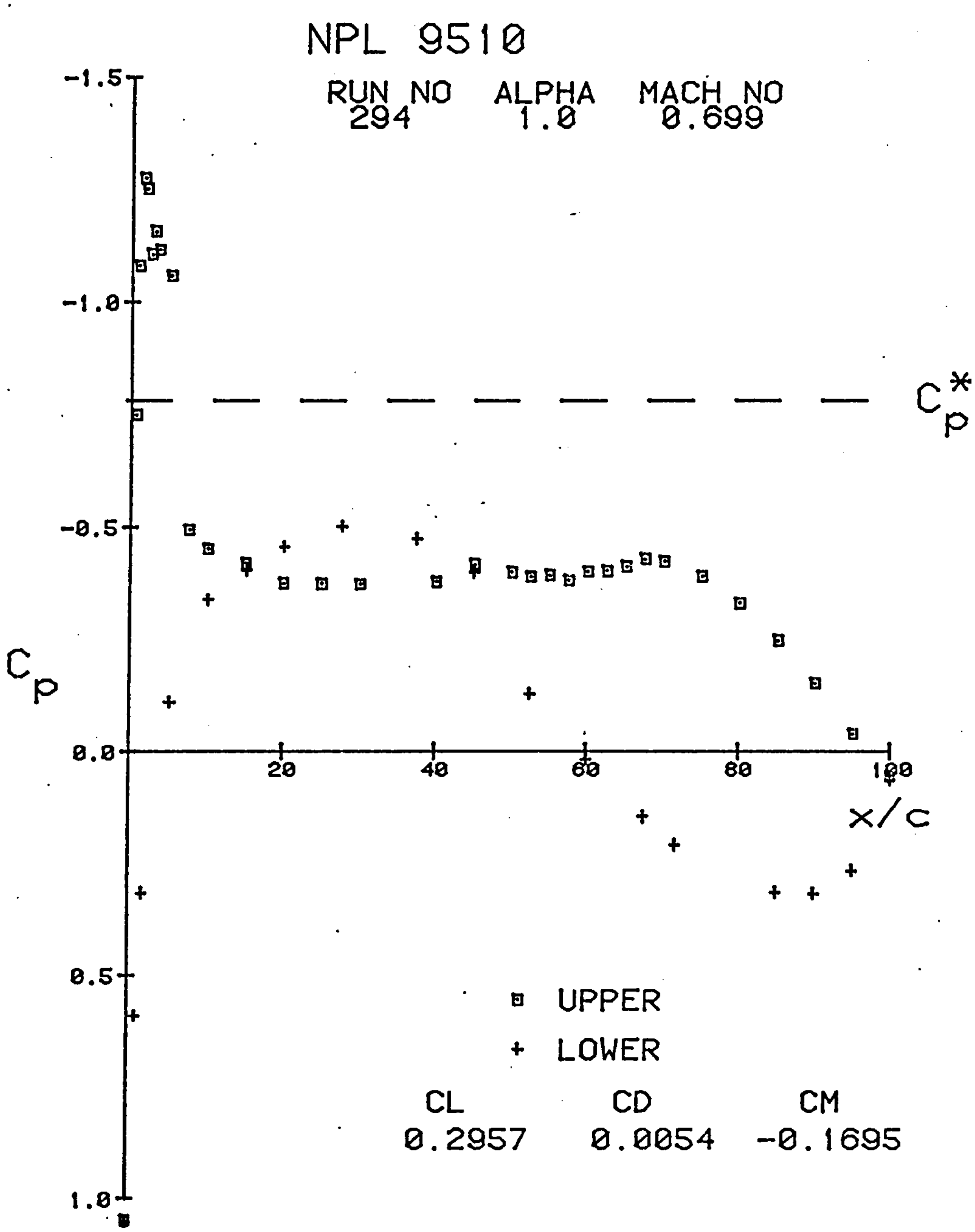

NPL 9510
RUN NO 294
ALPHA 1.0
MACH NO 0.699
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p^*
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.2957
CD 0.0054
CM -0.1695

Fig. 3.11.

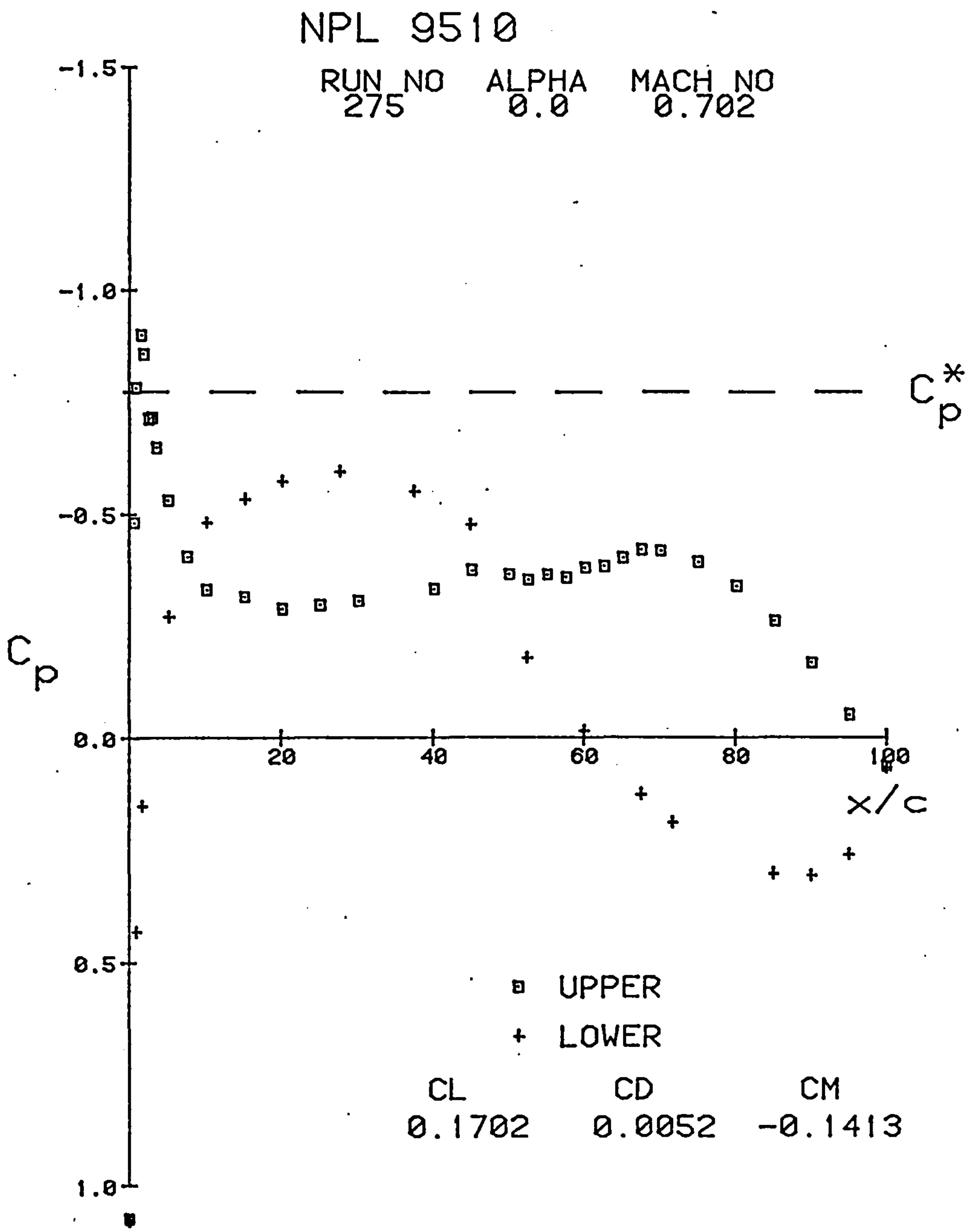
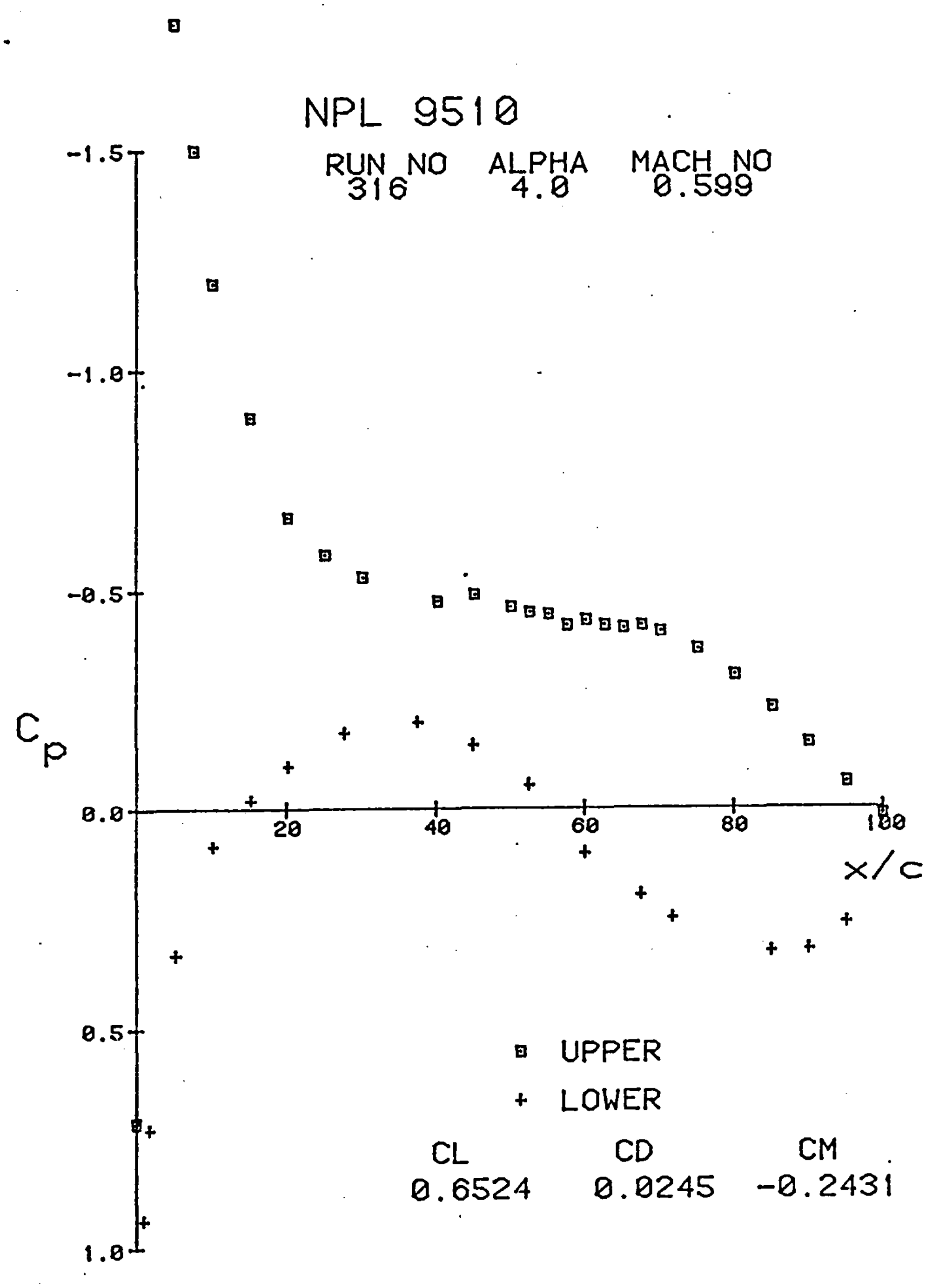

NPL 9510
RUN NO 275
ALPHA 0.0
MACH NO 0.702
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1702
CD 0.0052
CM -0.1413

Fig. 3.12.



NPL 9510
RUN NO 316
ALPHA 4.0
MACH NO 0.599
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6524
CD 0.0245
CM -0.2431

Fig. 3.13.

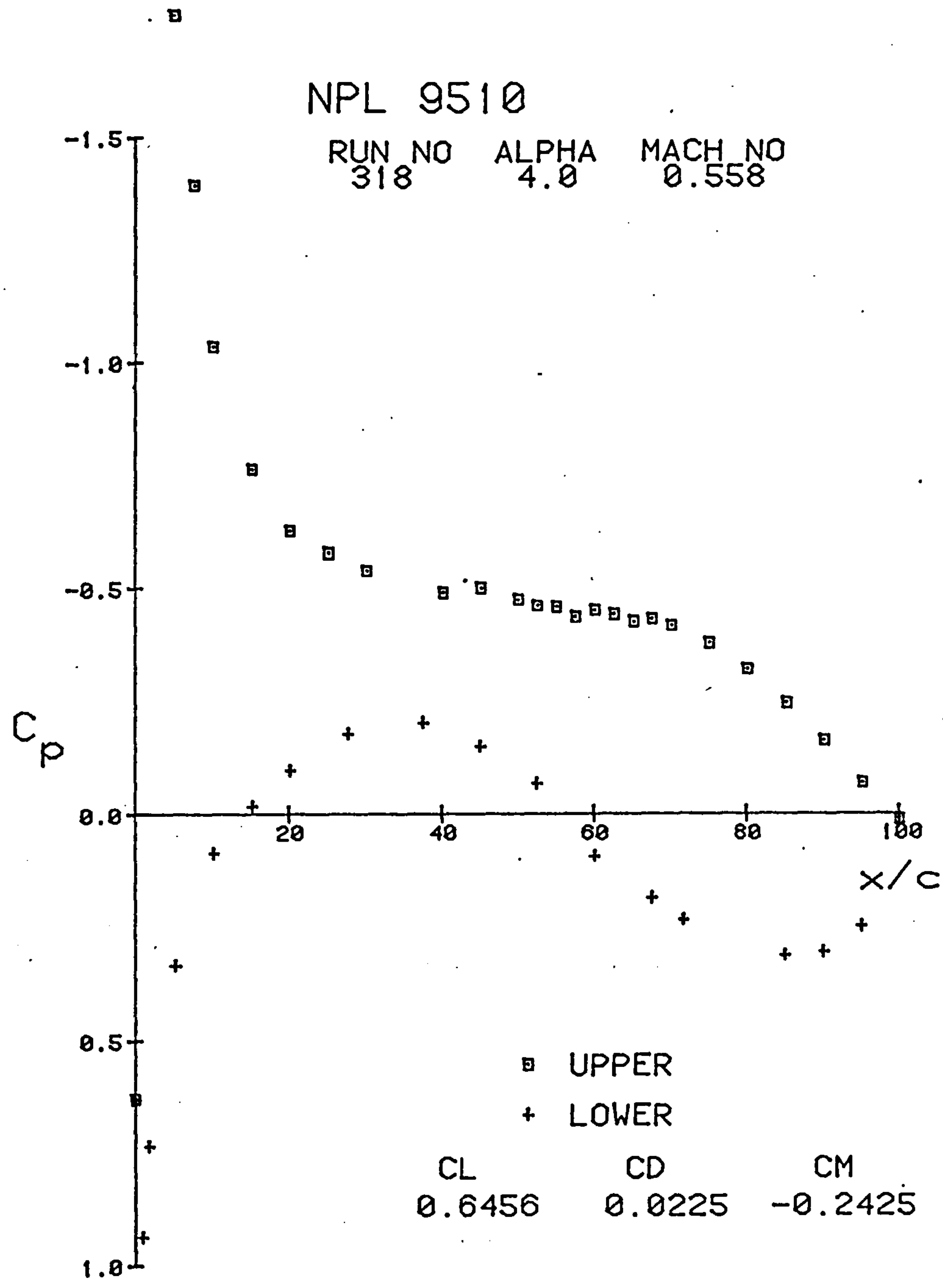
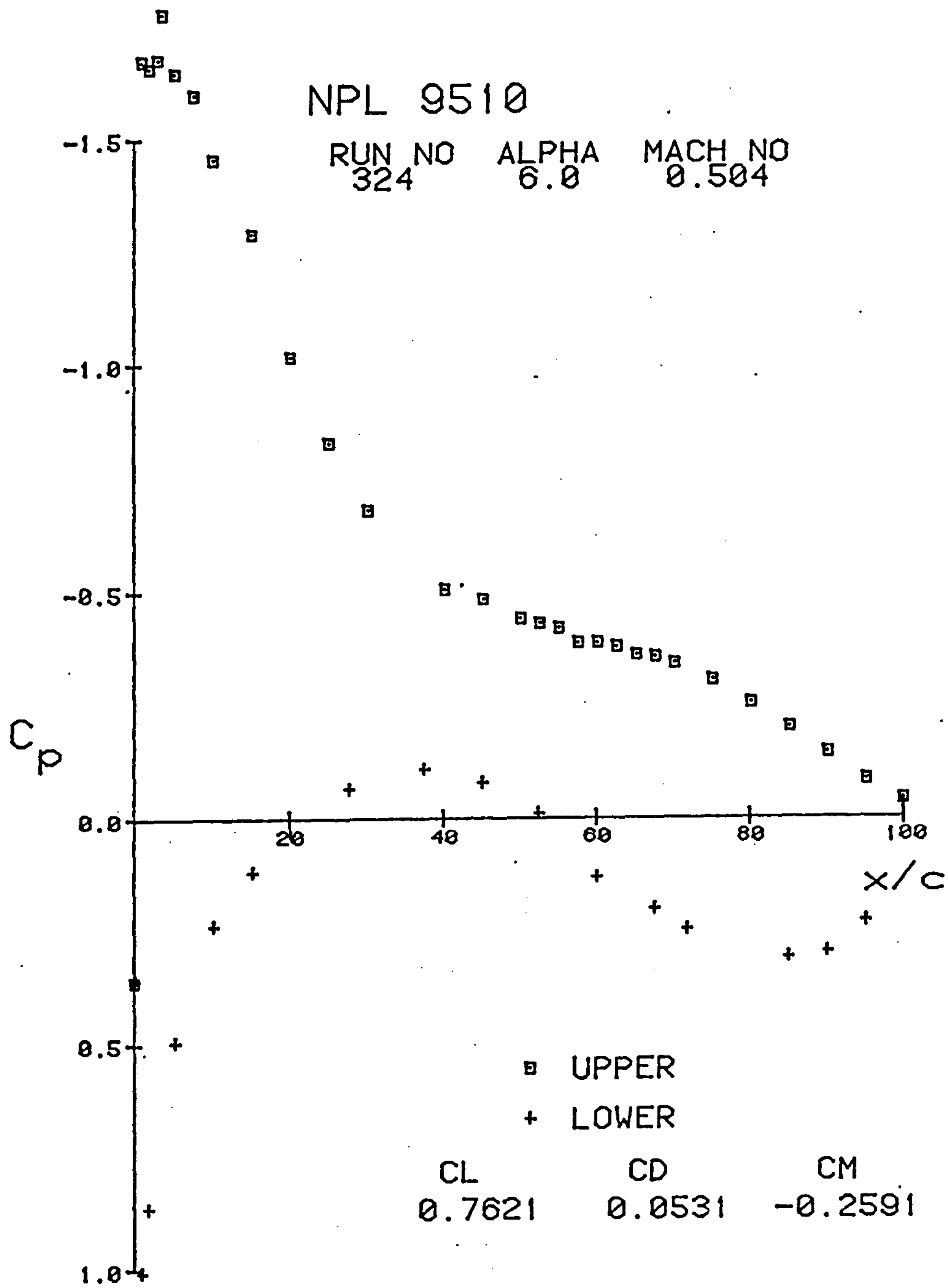

NPL 9510
RUN NO 318
ALPHA 4.0
MACH NO 0.558
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6456
CD 0.0225
CM -0.2425

Fig. 3.14.



NPL 9510
RUN NO 324
ALPHA 6.0
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 0.7621
CD 0.0531
CM -0.2591

Fig. 3.15.

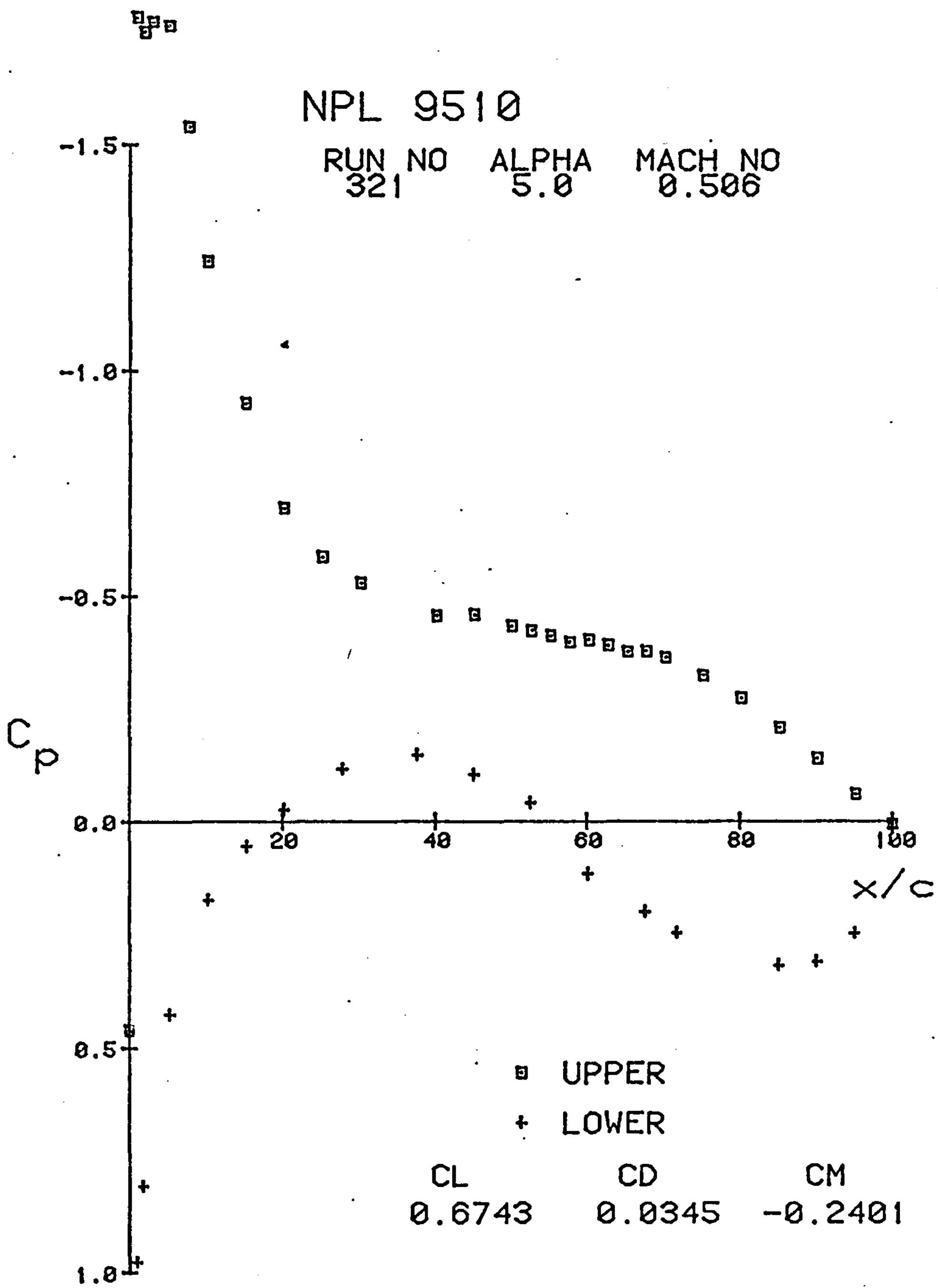

NPL 9510
RUN NO 321
ALPHA 5.0
MACH NO 0.506
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6743
CD 0.0345
CM -0.2401

Fig. 3.16.

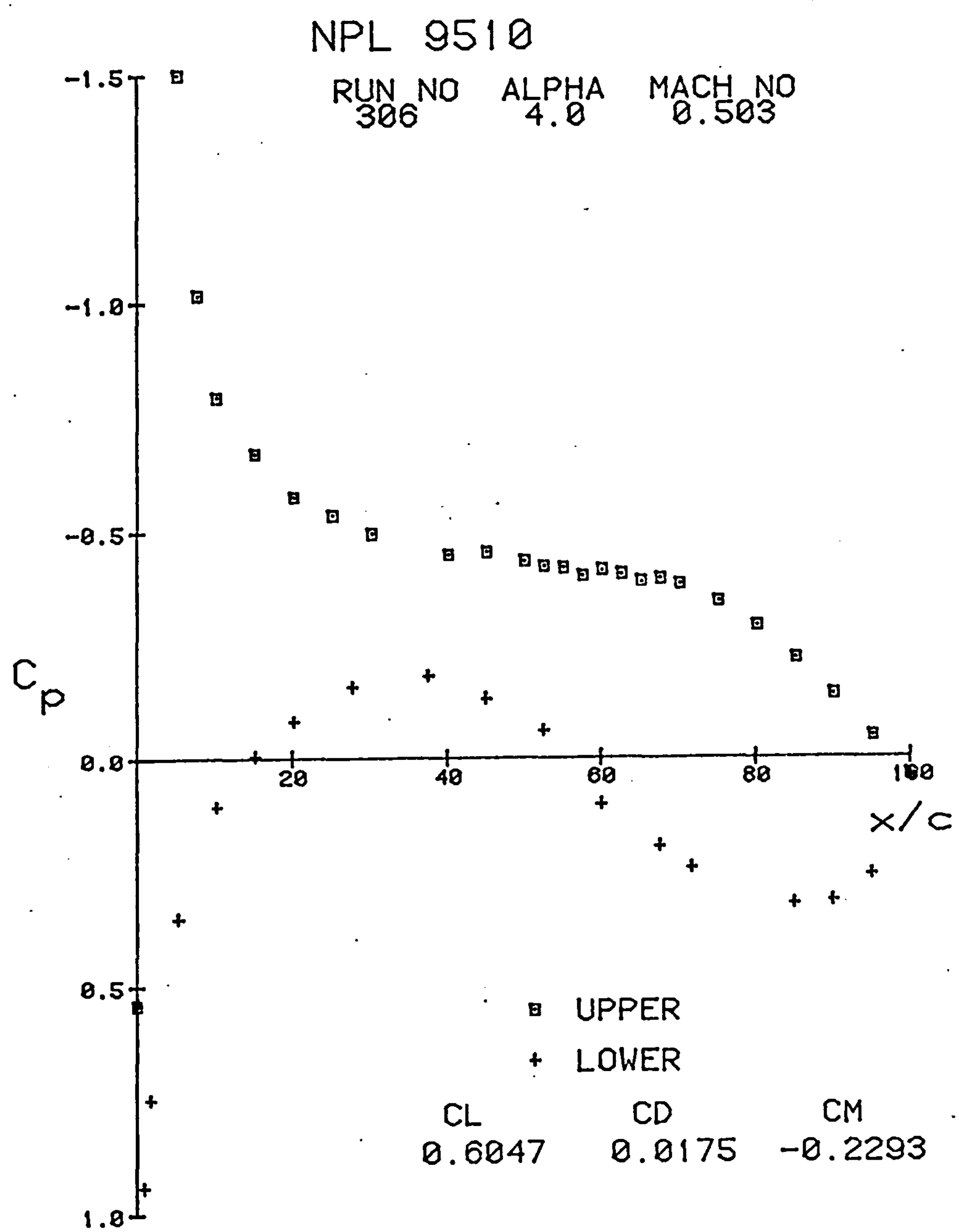

NPL 9510
RUN NO 306
ALPHA 4.0
MACH NO 0.503
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6047
CD 0.0175
CM -0.2293

Fig. 3.17.

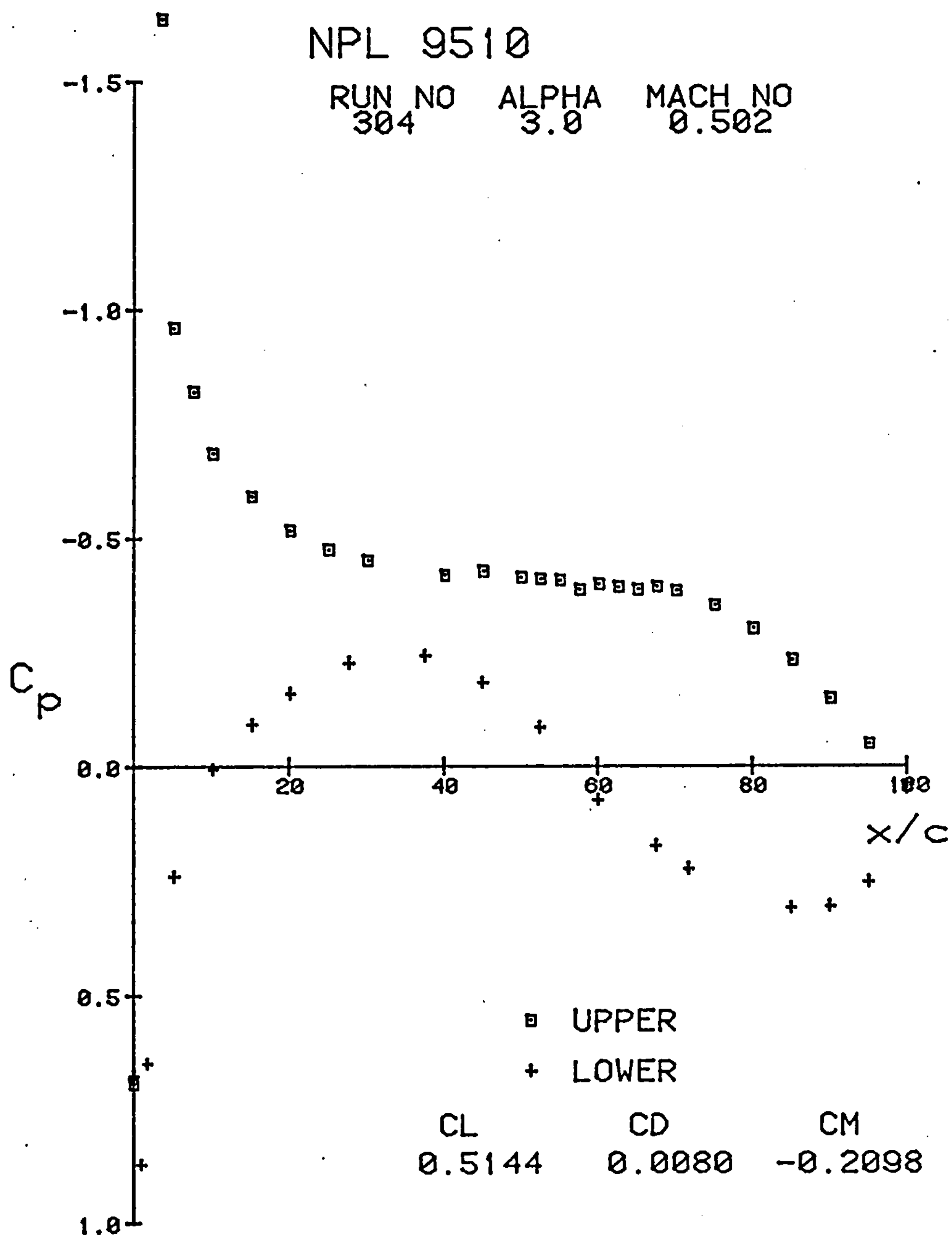

NPL 9510
RUN NO 304
ALPHA 3.0
MACH NO 0.502
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.5144
CD 0.0080
CM -0.2098

Fig. 3.18.

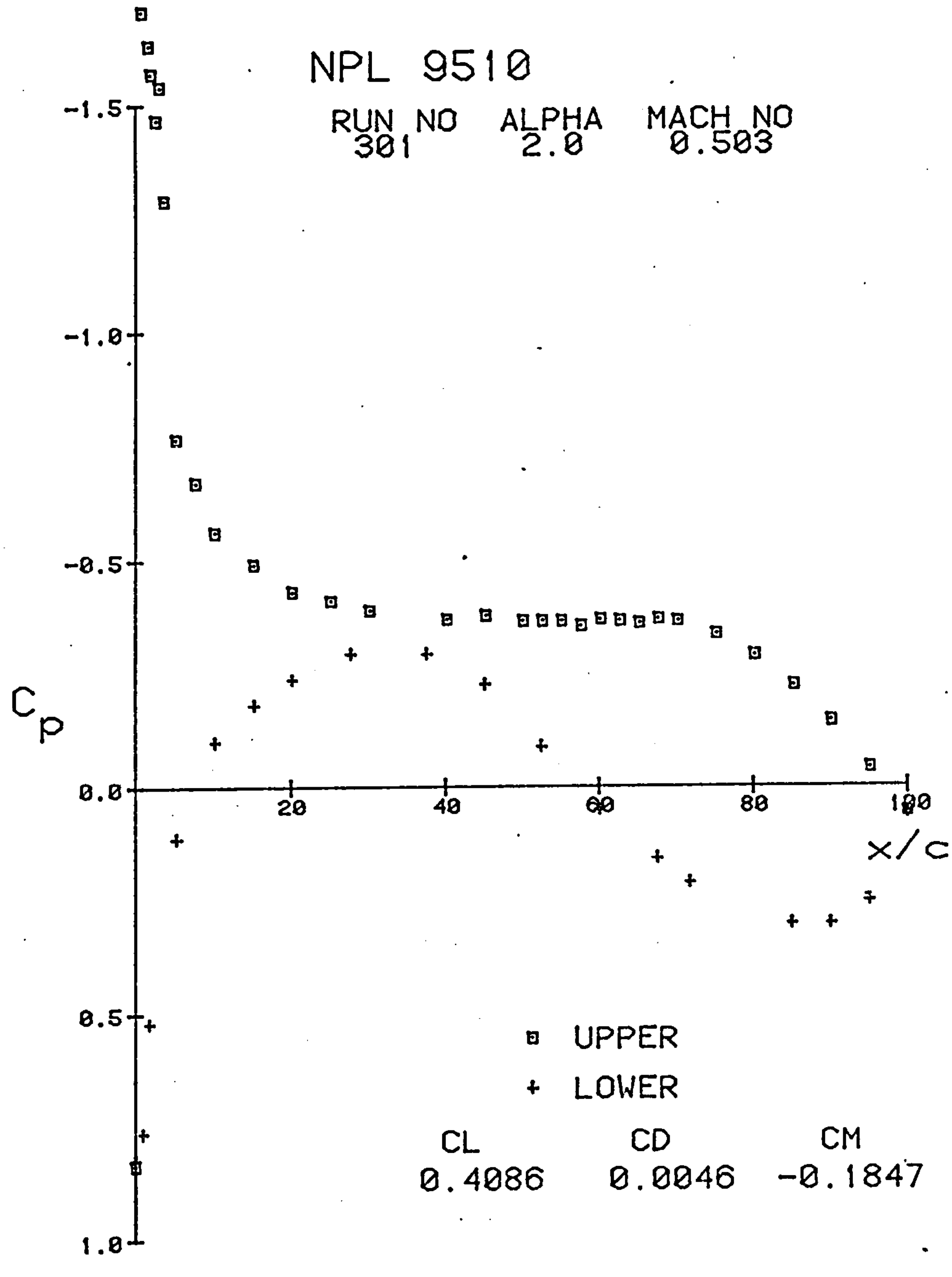

NPL 9510
RUN NO 301
ALPHA 2.0
MACH NO 0.503
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4086
CD 0.0046
CM -0.1847

Fig. 3.19.

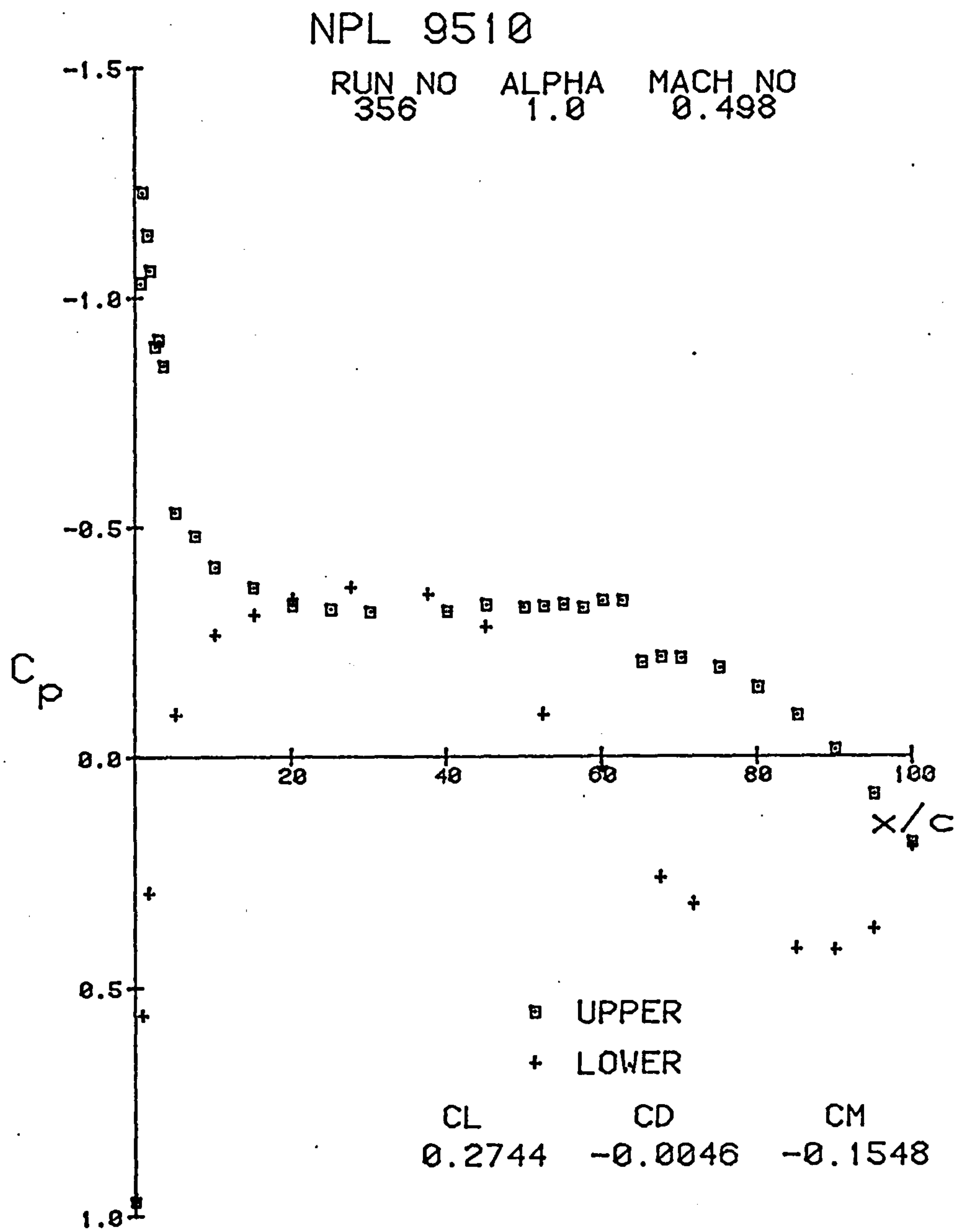

NPL 9510
RUN NO 356
ALPHA 1.0
MACH NO 0.498
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
X/C
UPPER
LOWER
CL 0.2744
CD -0.0046
CM -0.1548

Fig. 3.20.

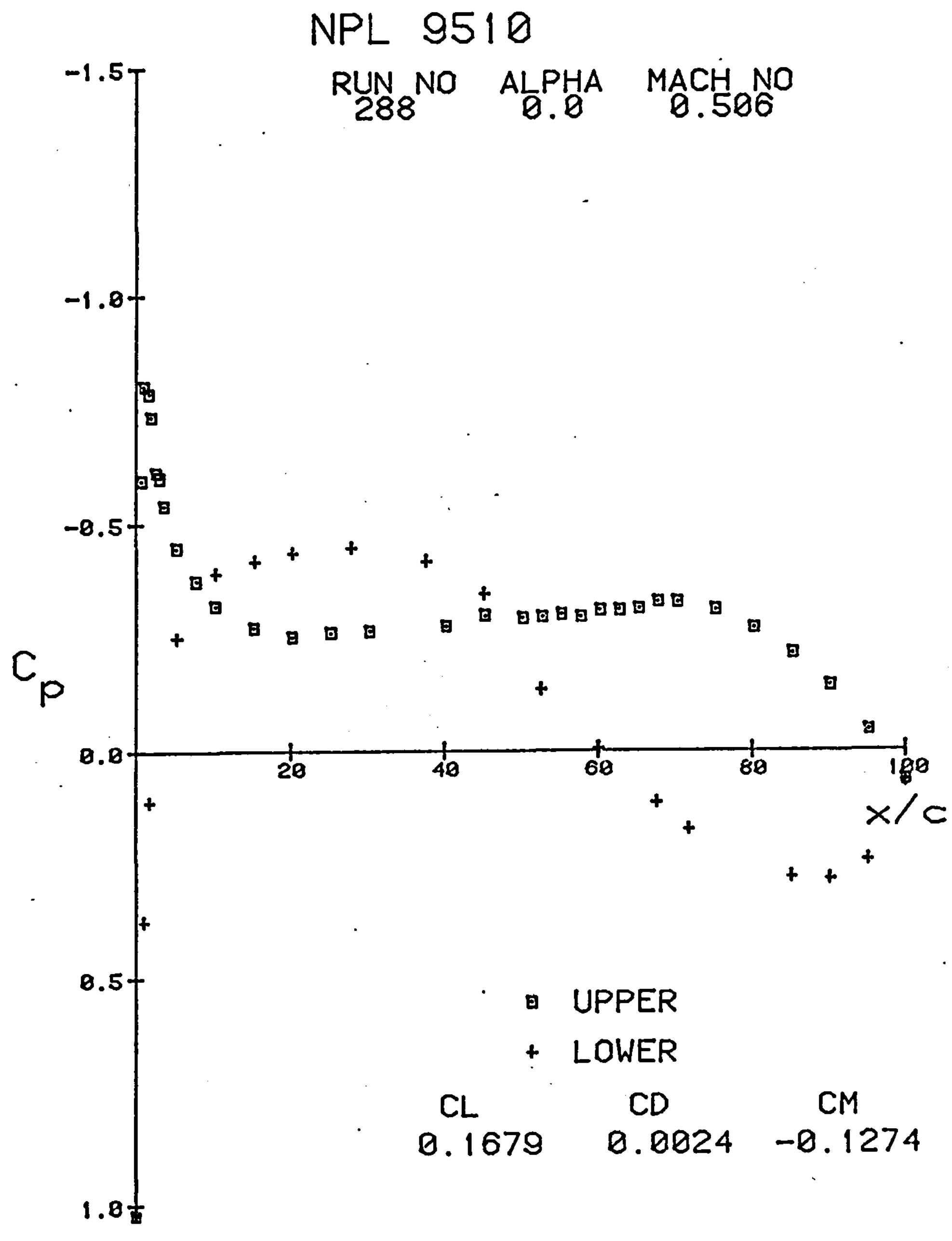

NPL 9510
RUN NO 288
ALPHA 0.0
MACH NO 0.506
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1679
CD 0.0024
CM -0.1274

Fig. 3.21.

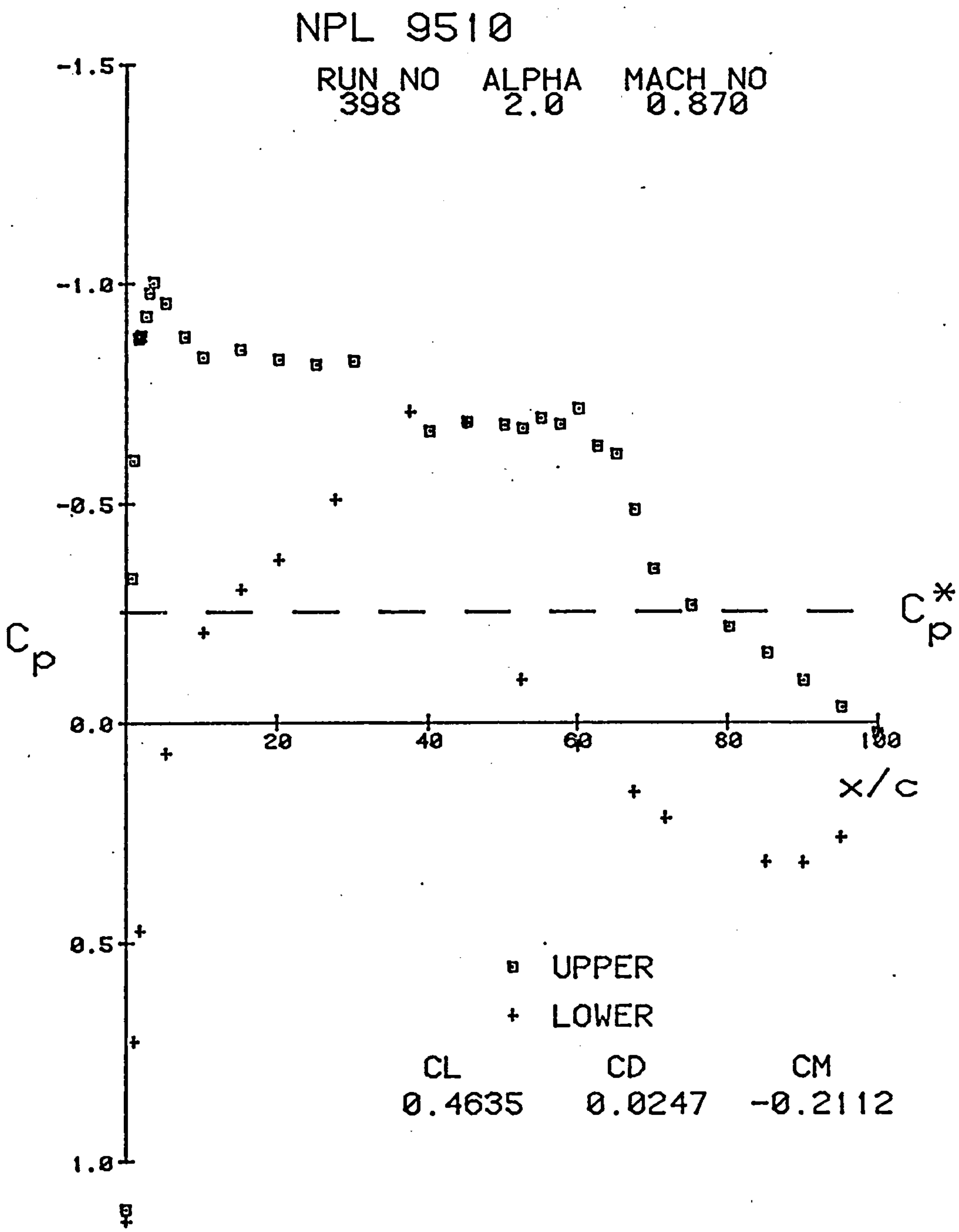

NPL 9510
RUN NO 398
ALPHA 2.0
MACH NO 0.870
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4635
CD 0.0247
CM -0.2112

Fig. 3.22.

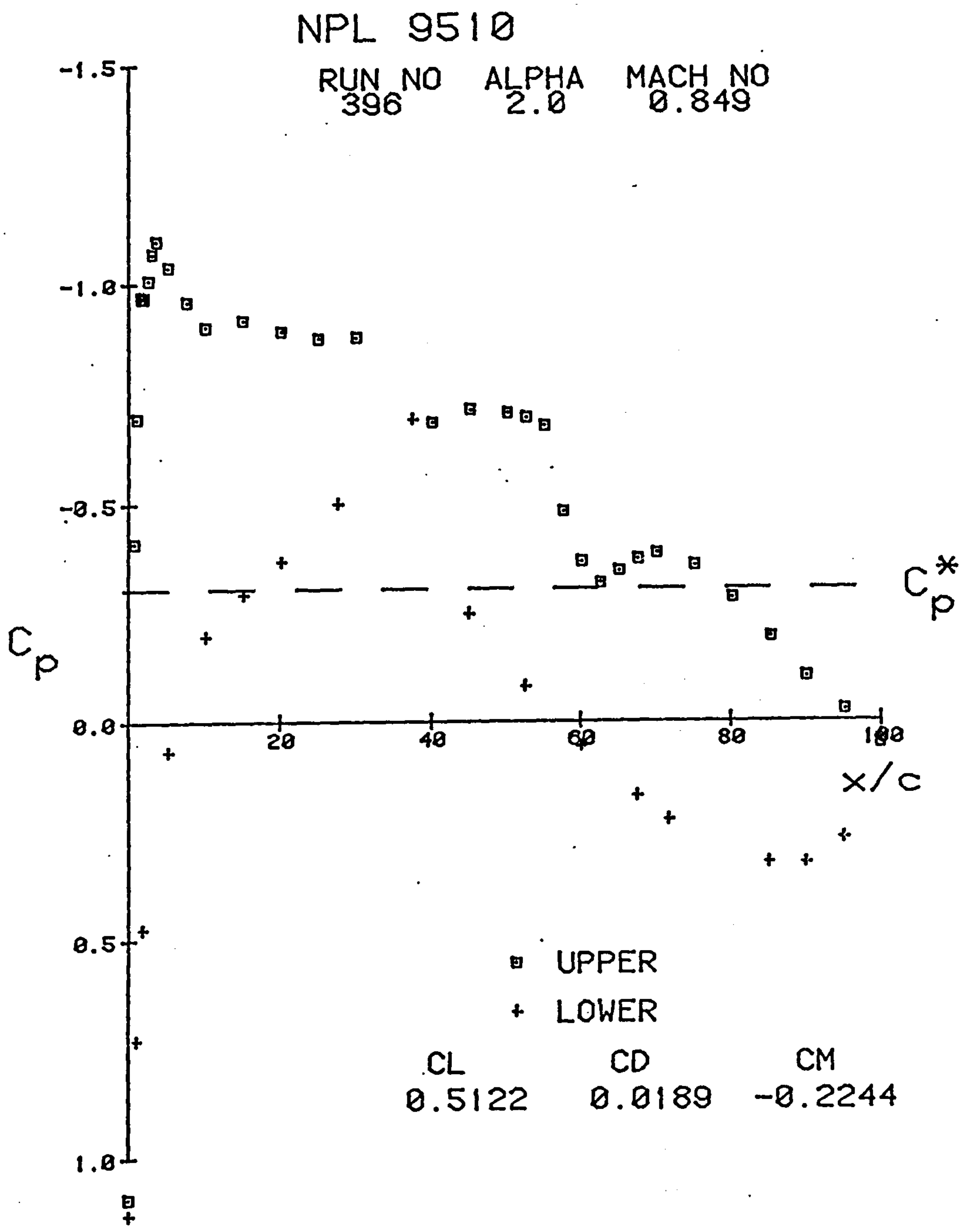
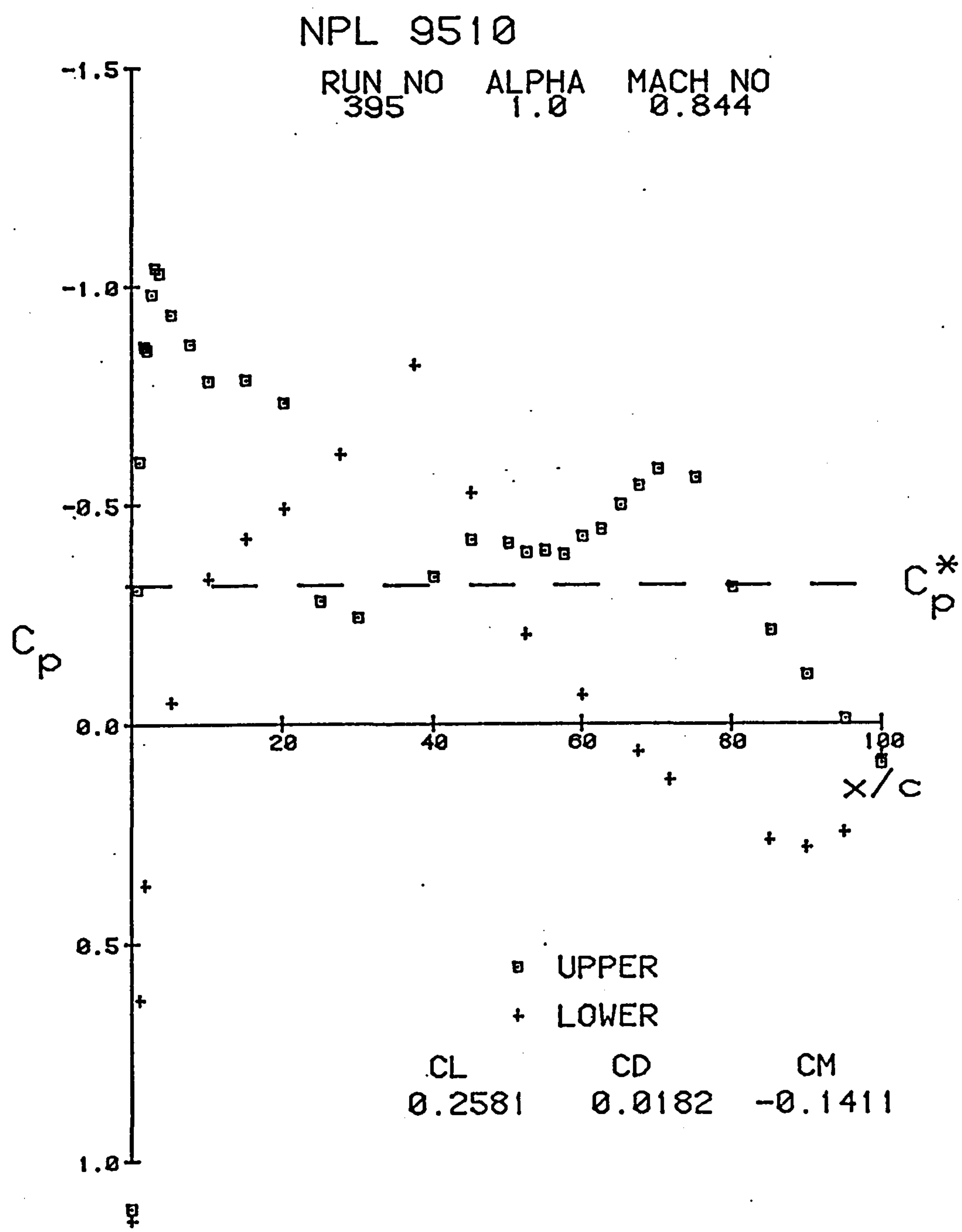

NPL 9510
RUN NO 396
ALPHA 2.0
MACH NO 0.849
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
x/c
Cp
Cp*
UPPER
LOWER
CL 0.5122
CD 0.0189
CM -0.2244

Fig. 3.23.



NPL 9510
RUN NO 395
ALPHA 1.0
MACH NO 0.844
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
Cp
Cp*
x/c
UPPER
LOWER
CL 0.2581
CD 0.0182
CM -0.1411

Fig. 3.24.

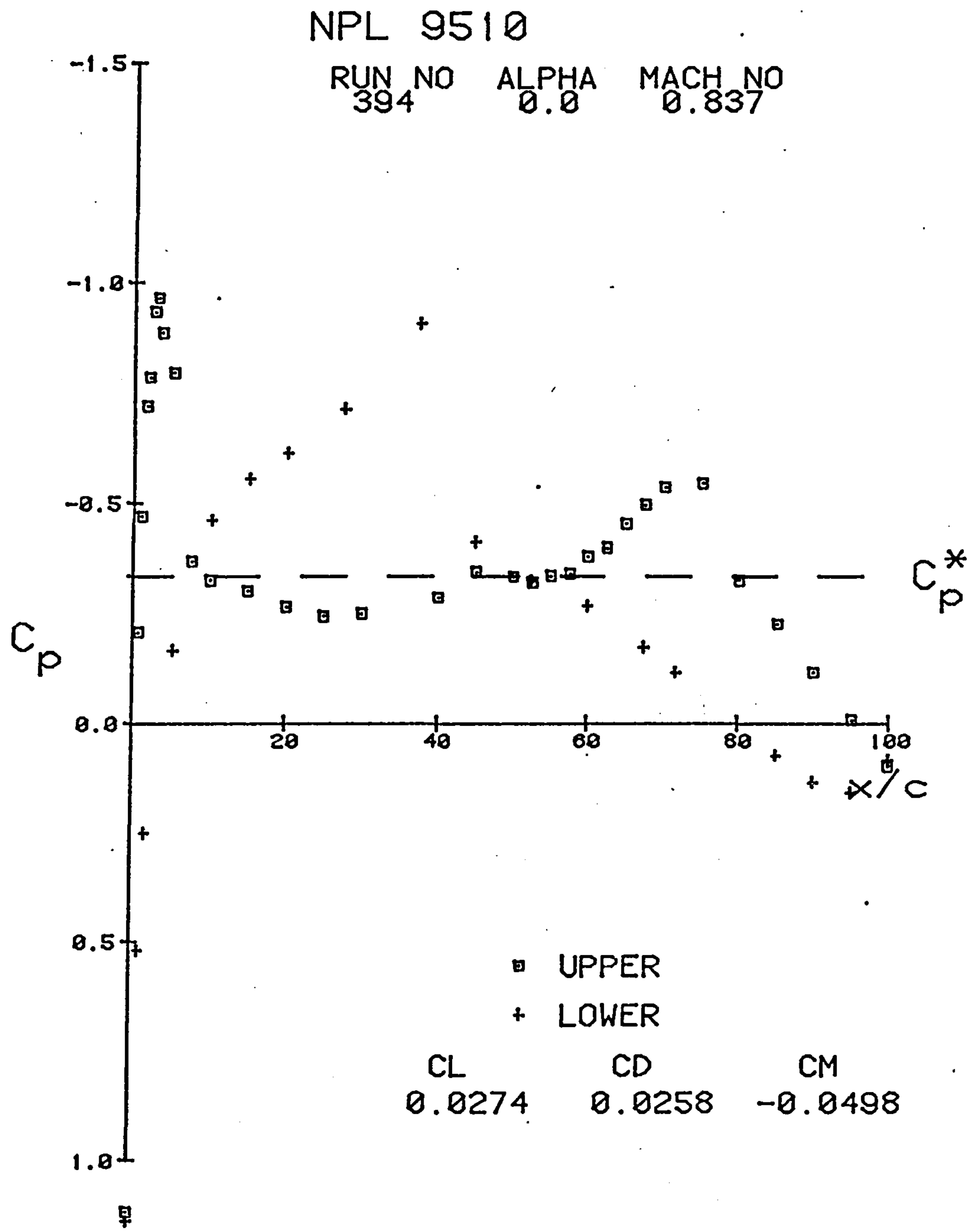

NPL 9510
RUN NO 394
ALPHA 0.0
MACH NO 0.837
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
Cp*
UPPER
LOWER
CL 0.0274
CD 0.0258
CM -0.0498

Fig. 3.25.

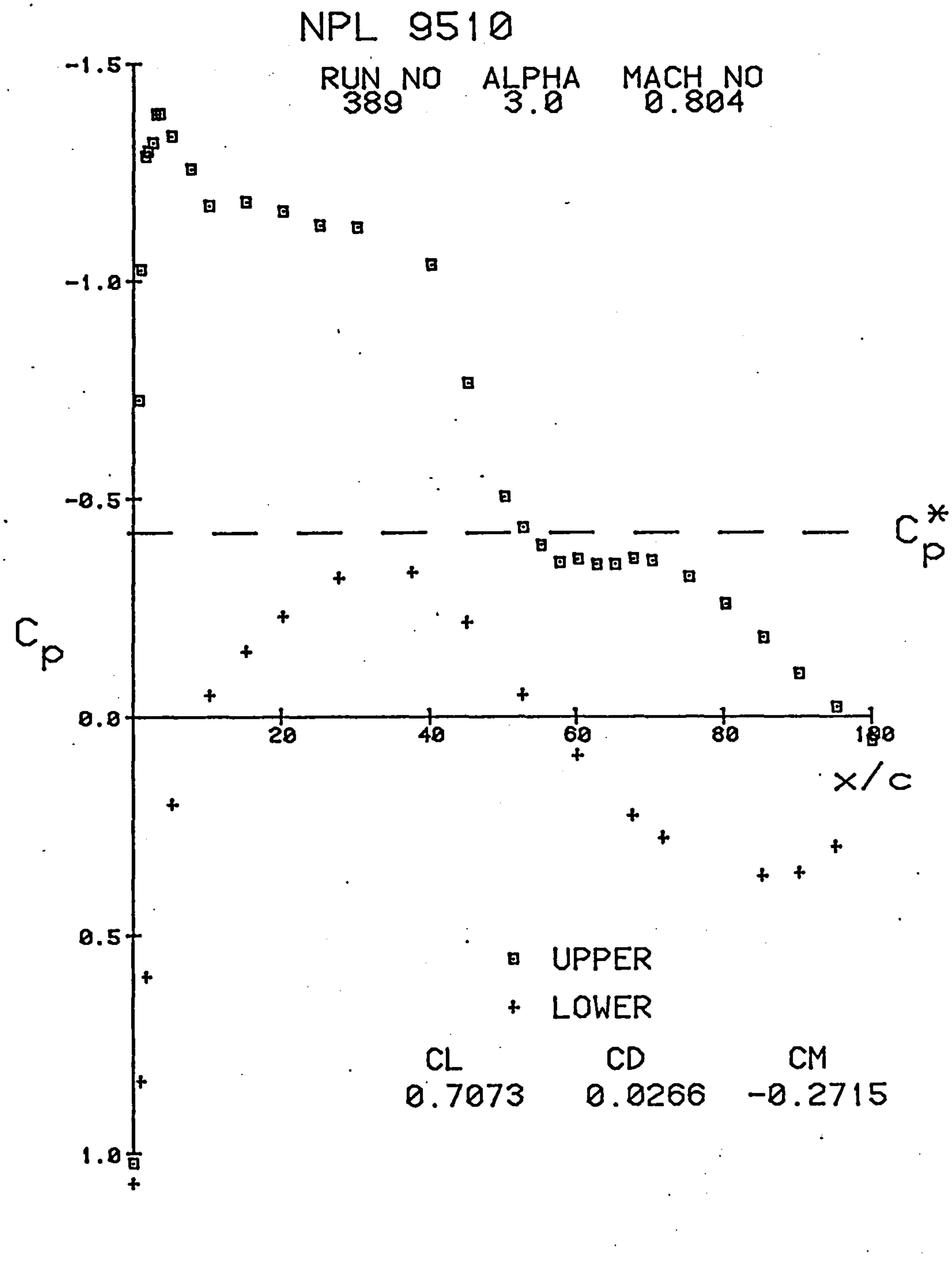
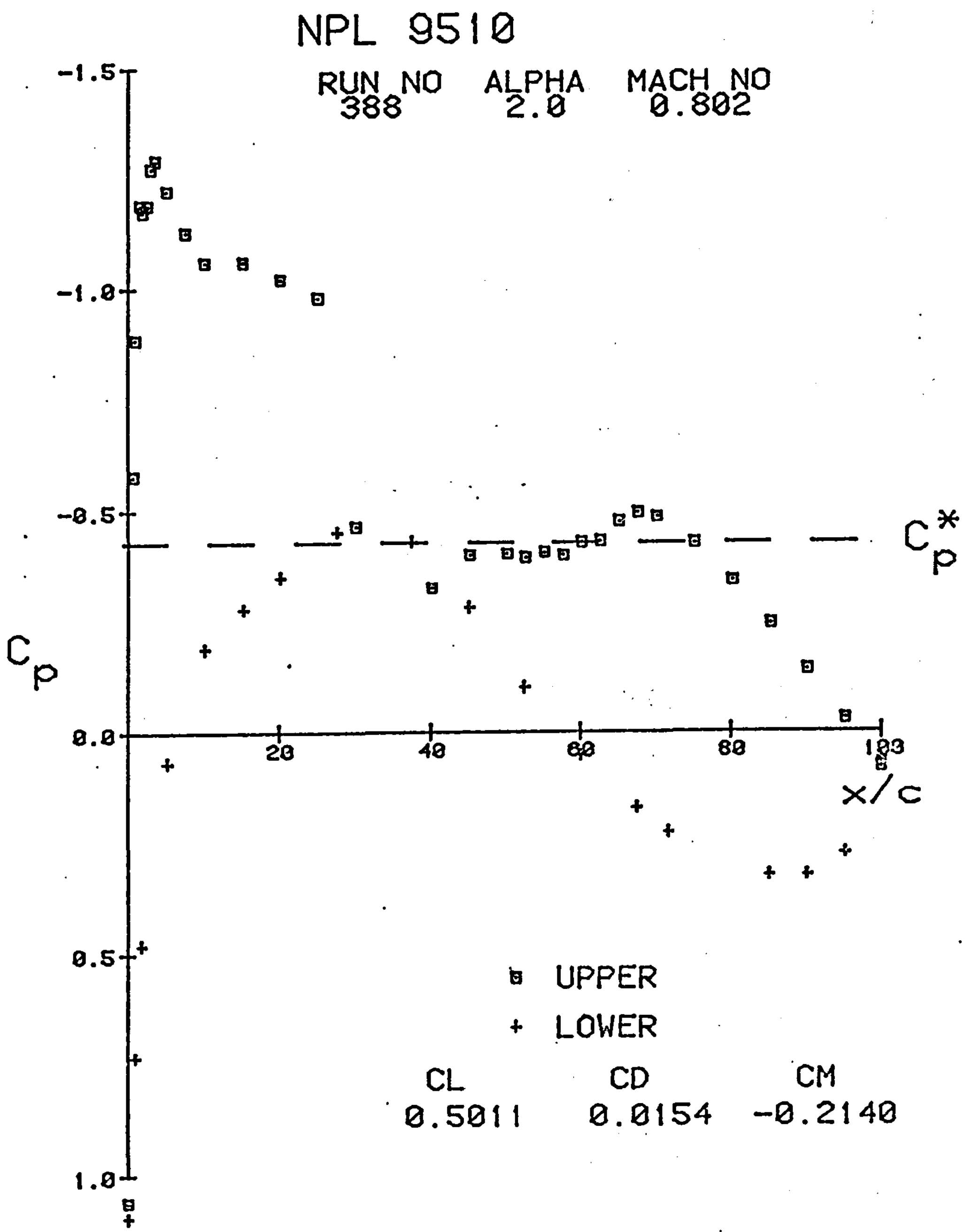

NPL 9510
RUN NO 389
ALPHA 3.0
MACH NO 0.804
-1.5
-1.0
-0.5
0.0
0.5
1.0
C_p
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.7073
CD 0.0266
CM -0.2715

Fig. 3.26.



NPL 9510
RUN NO 388
ALPHA 2.0
MACH NO 0.802
-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 0.5011
CD 0.0154
CM -0.2140

Fig. 3.27.

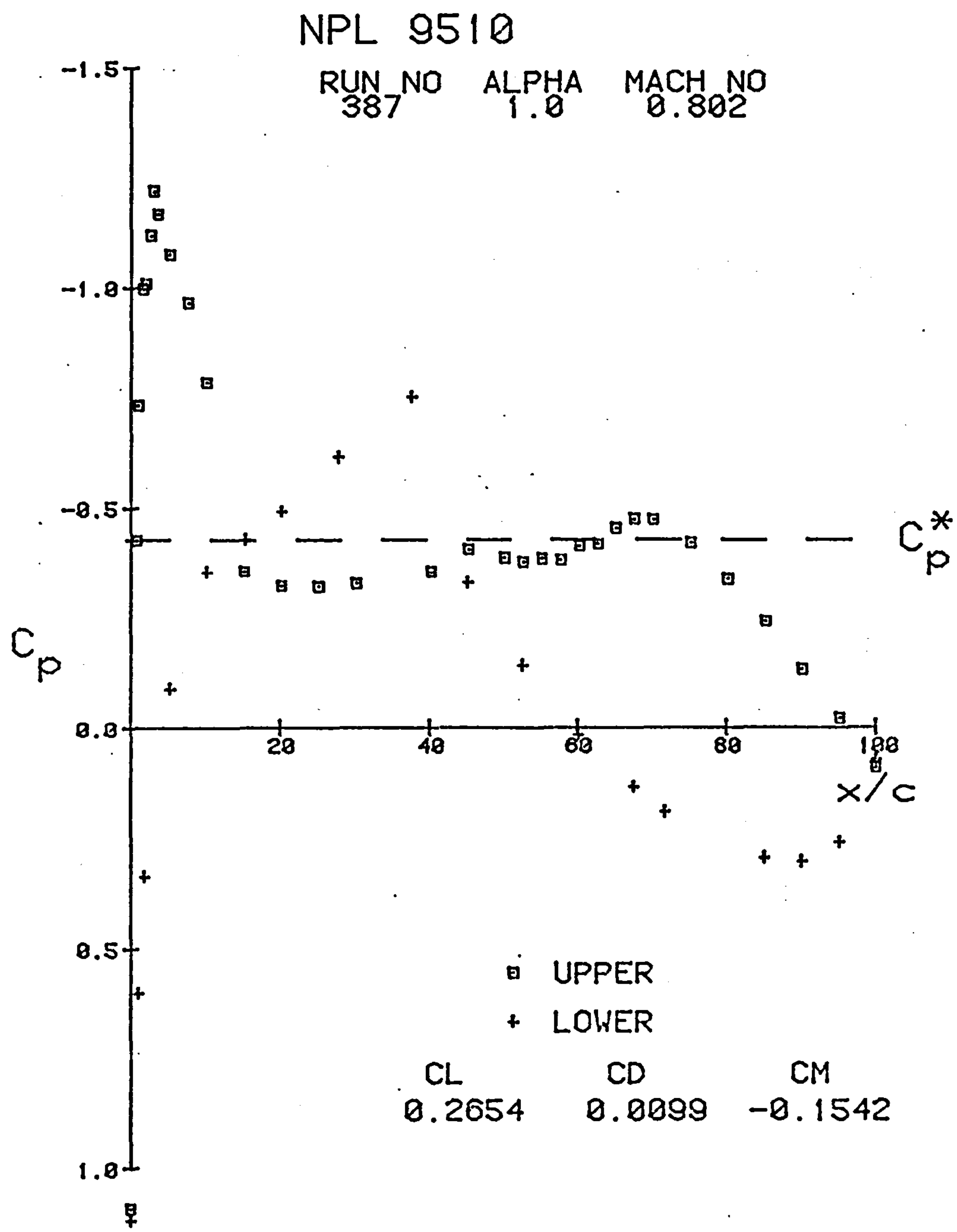

NPL 9510
RUN NO 387
ALPHA 1.0
MACH NO 0.802
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
Cp*
UPPER
LOWER
CL 0.2654
CD 0.0099
CM -0.1542

Fig. 3.28.

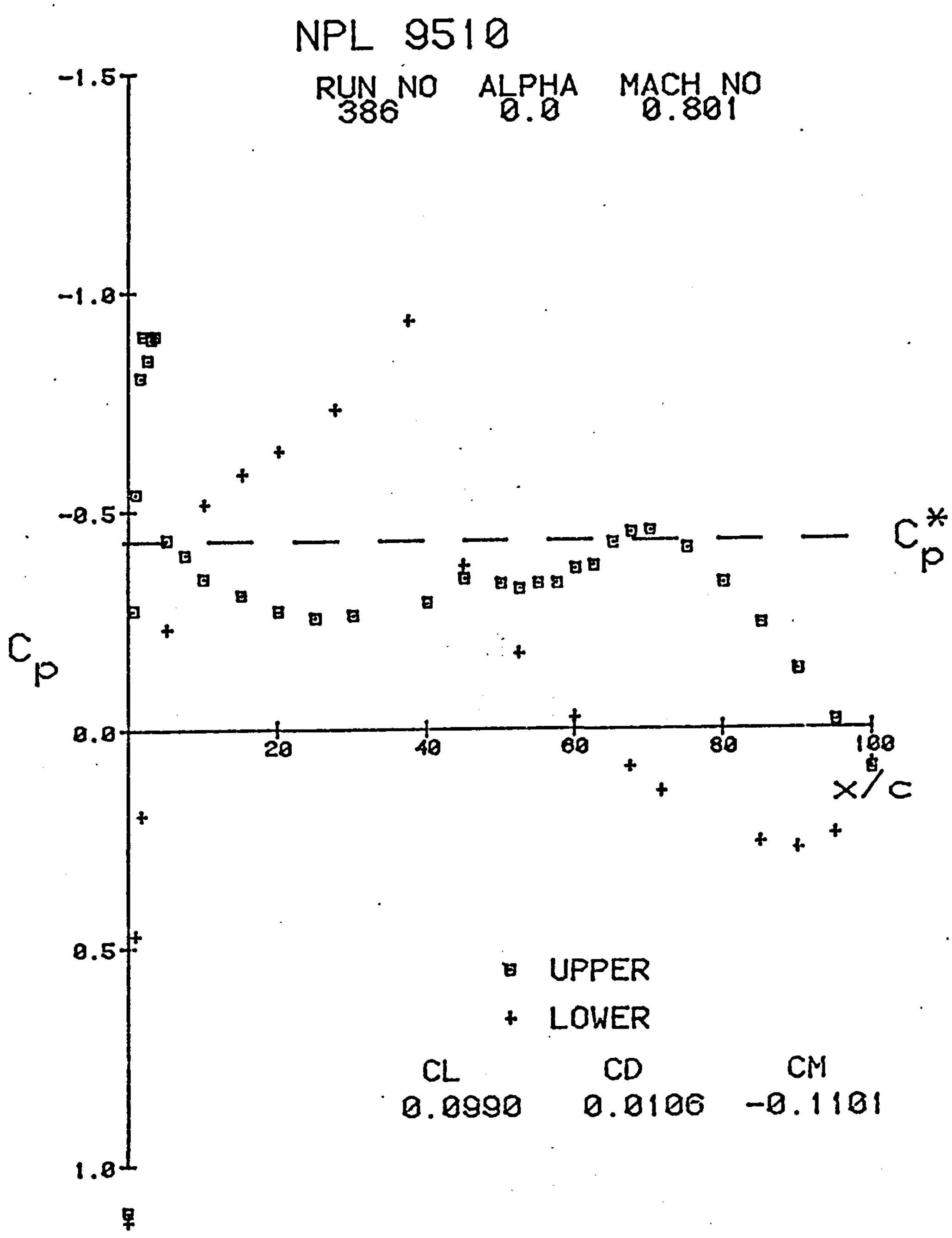
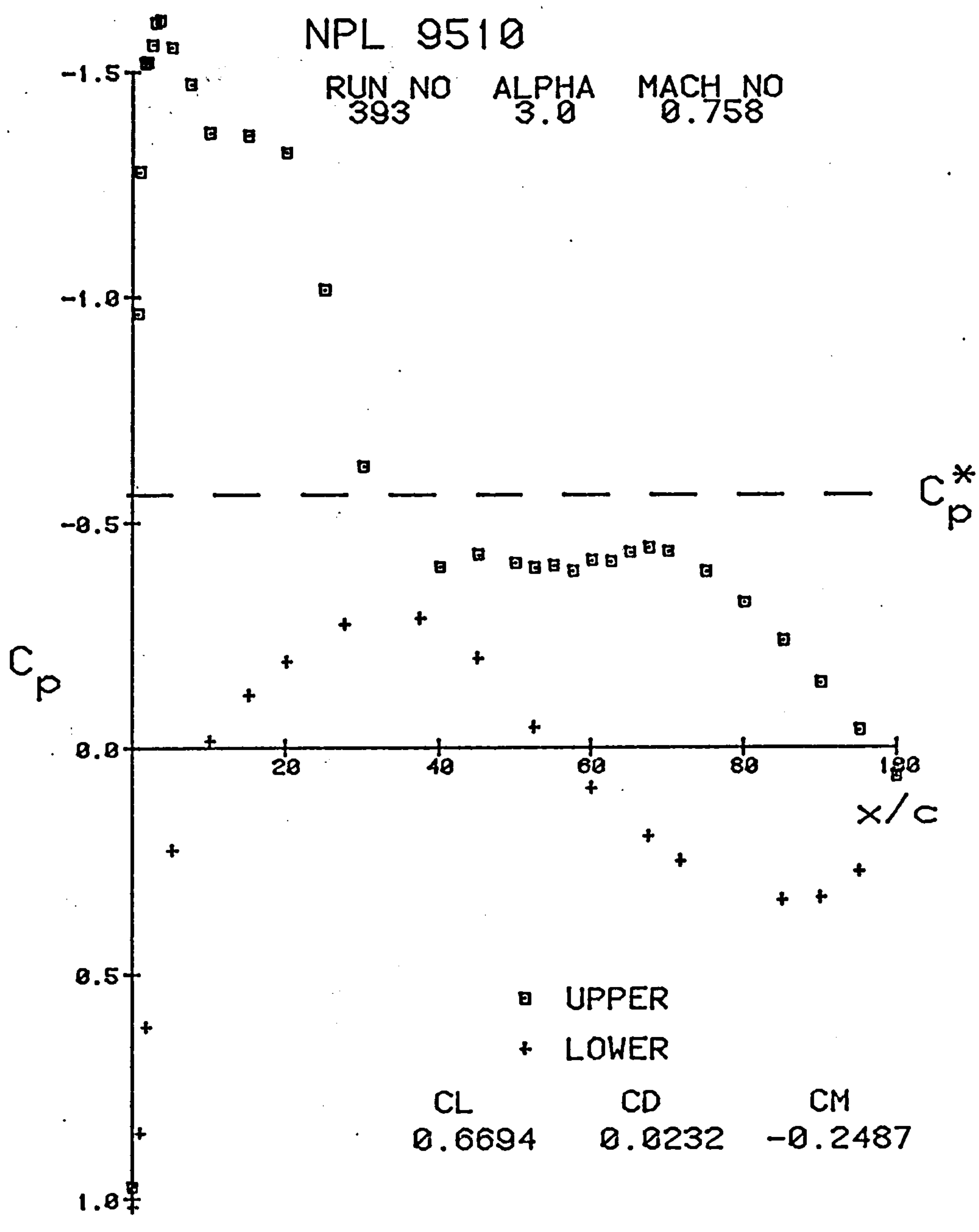

NPL 9510
RUN NO 386
ALPHA 0.0
MACH NO 0.801
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
Cp
Cp*
x/c
UPPER
LOWER
CL 0.0990
CD 0.0106
CM -0.1101

Fig. 3.29.



NPL 9510
RUN NO 393
ALPHA 3.0
MACH NO 0.758
-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 0.6694
CD 0.0232
CM -0.2487

Fig. 3.30.

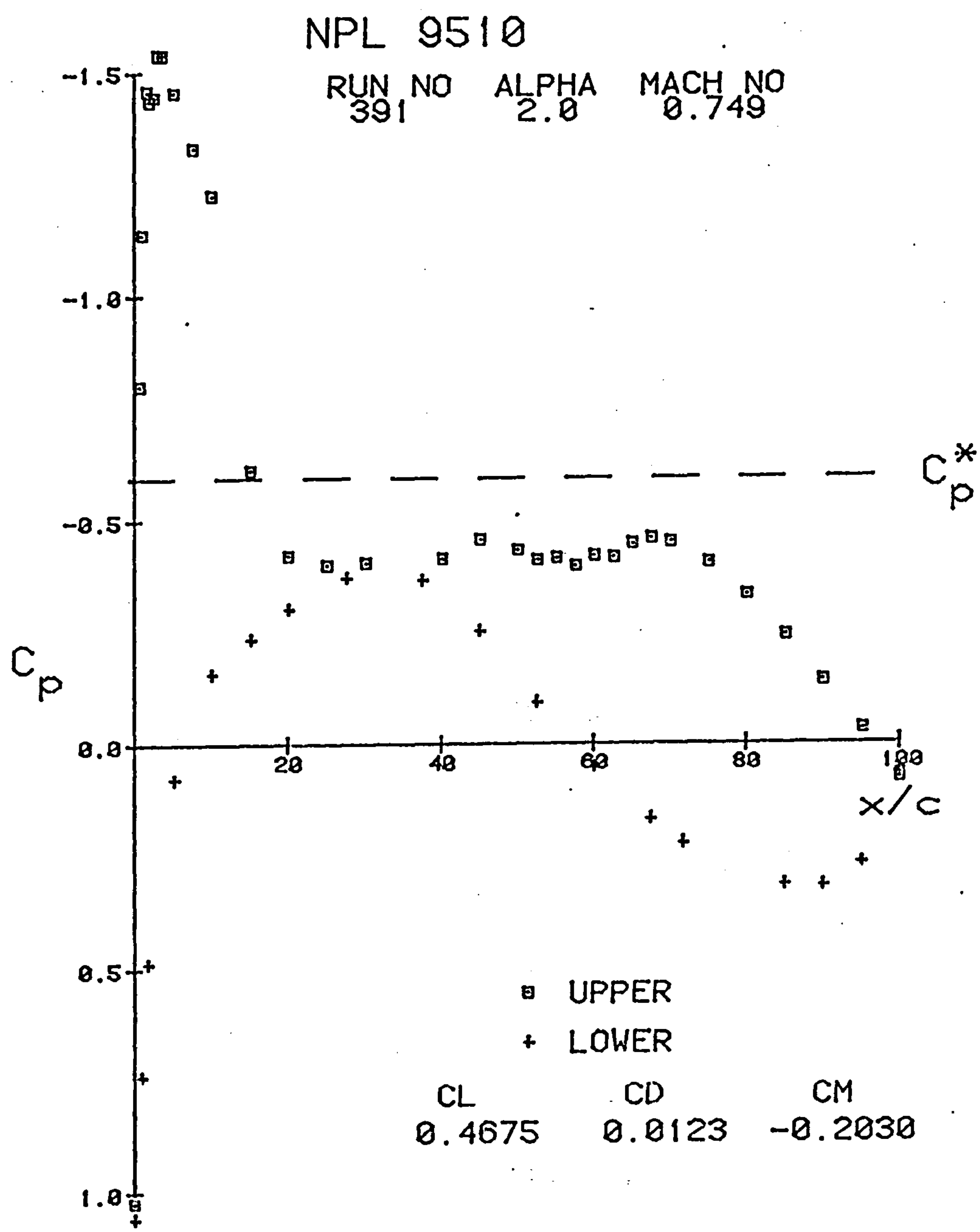
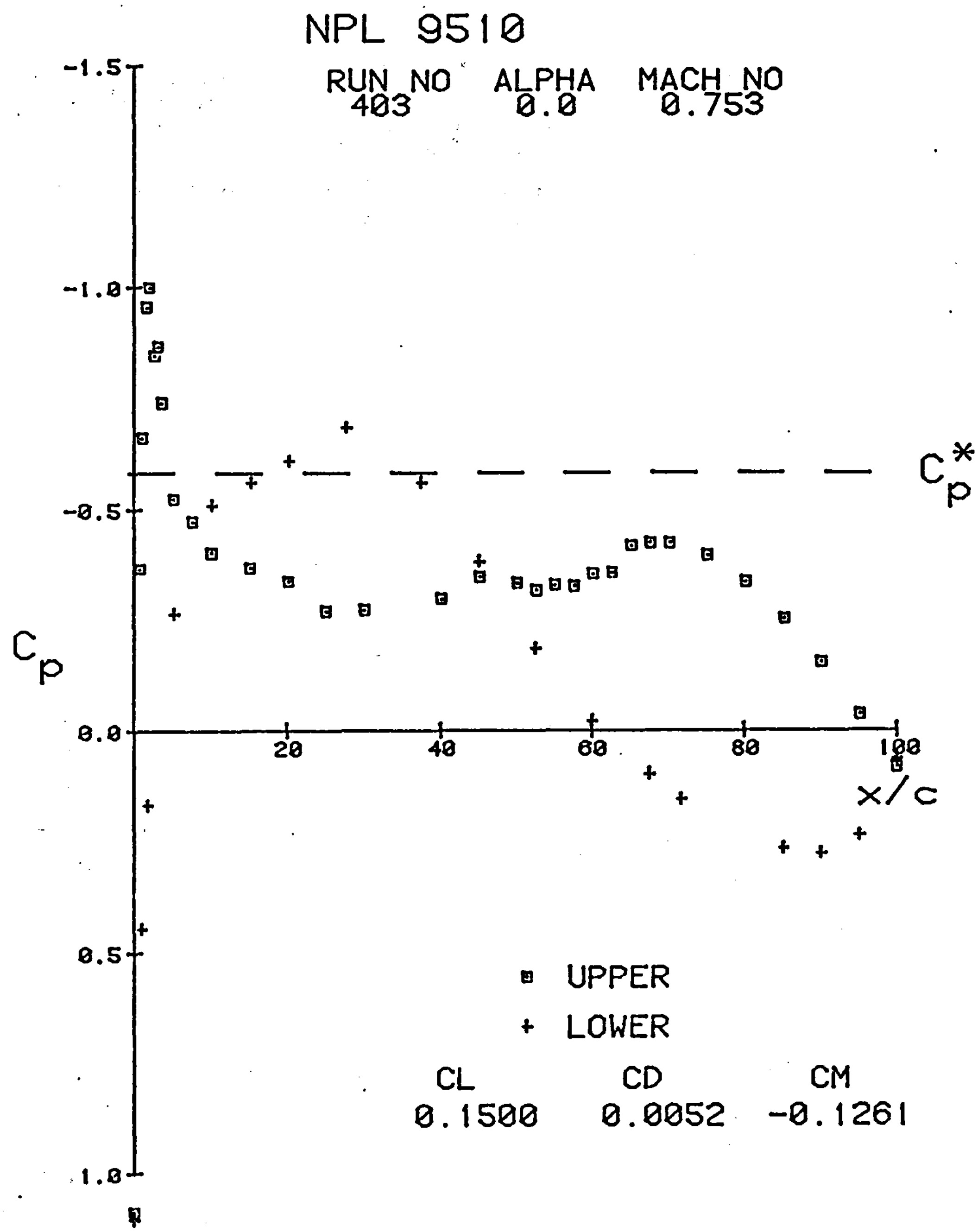

NPL 9510
RUN NO 391
ALPHA 2.0
MACH NO 0.749
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
C_p^*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4675
CD 0.0123
CM -0.2030

Fig. 3.31.



NPL 9510
RUN NO 403
ALPHA 0.0
MACH NO 0.753
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1500
CD 0.0052
CM -0.1261

Fig. 3.32.

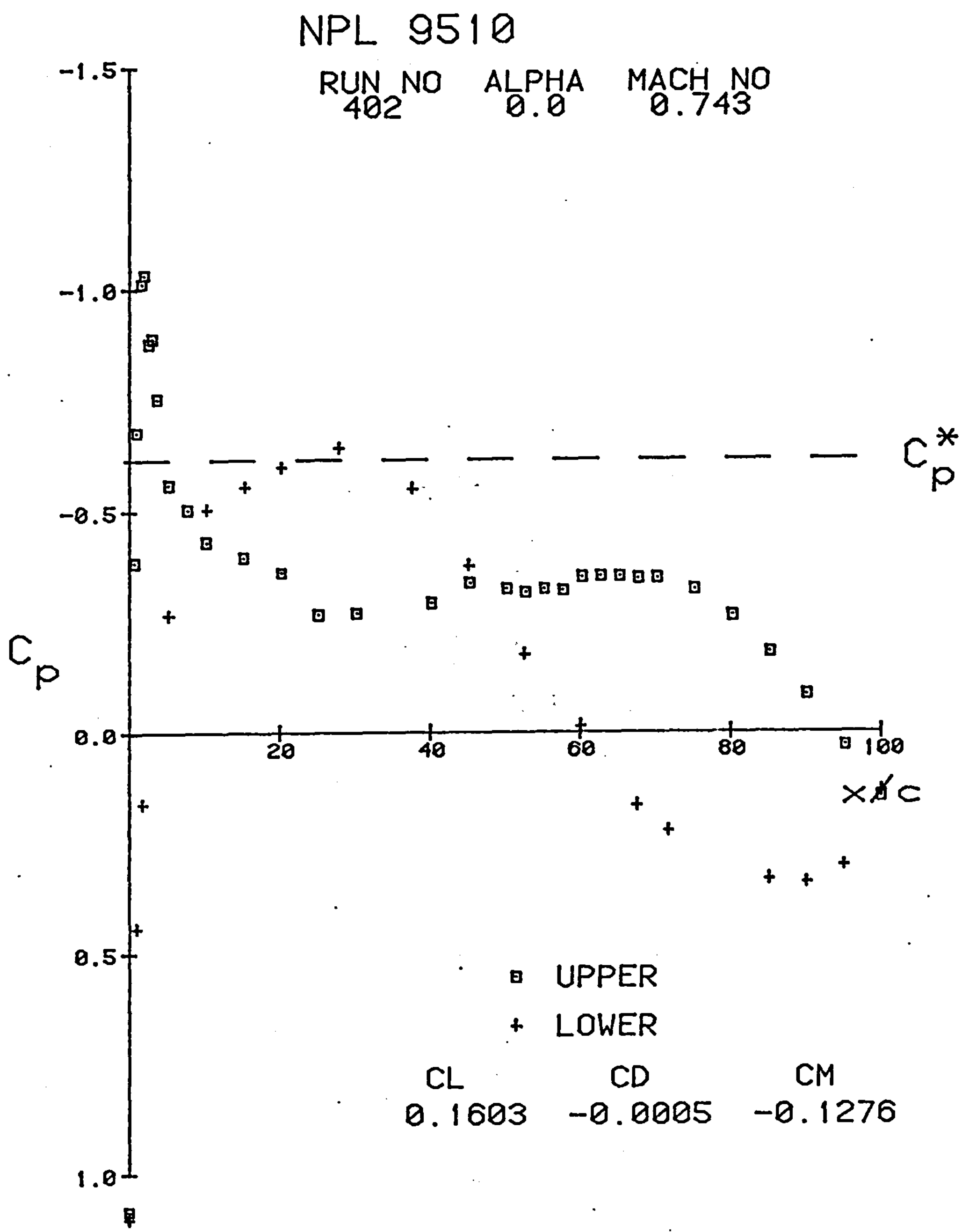

NPL 9510
RUN NO 402
ALPHA 0.0
MACH NO 0.743
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1603
CD -0.0005
CM -0.1276

Fig. 3.33.

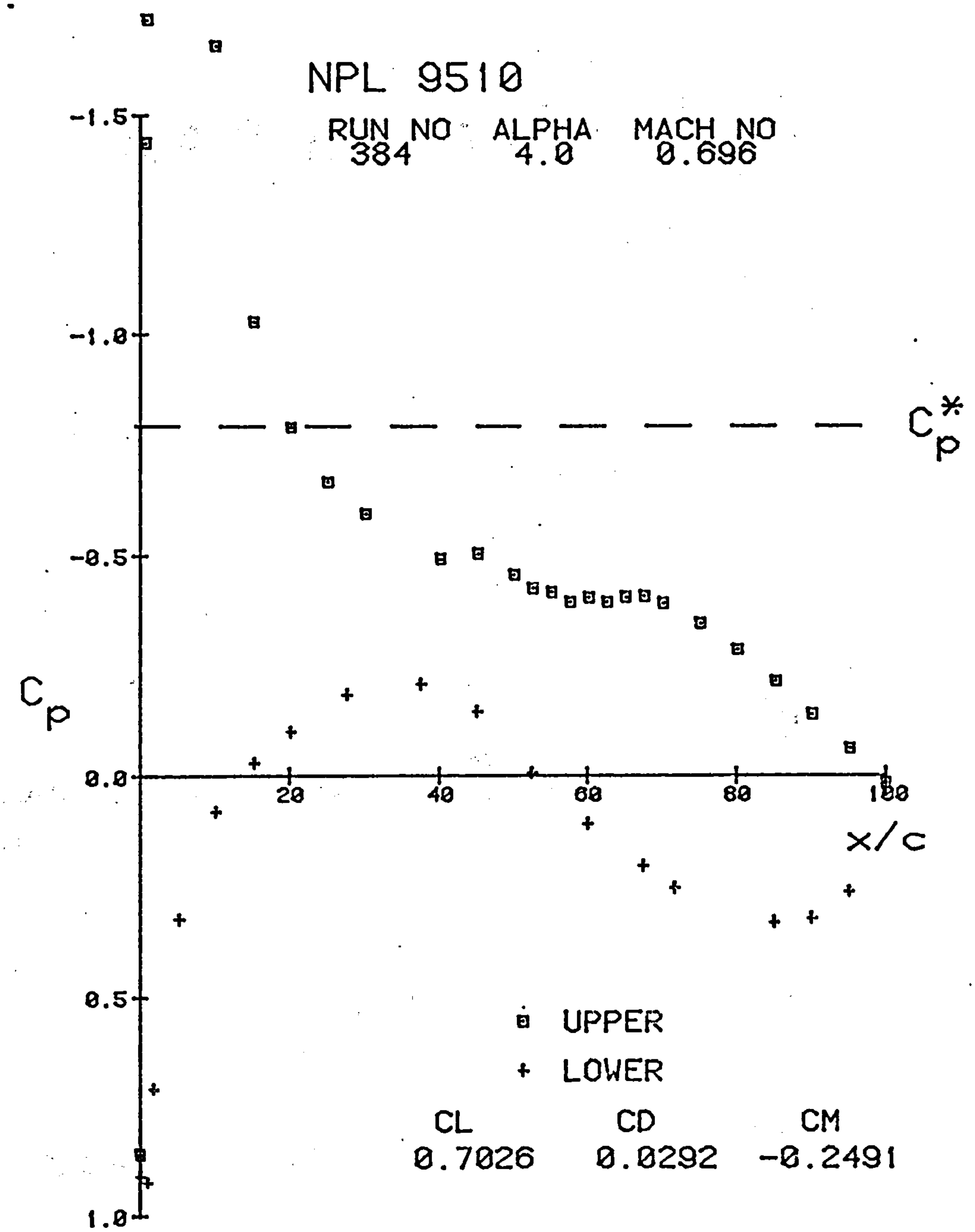
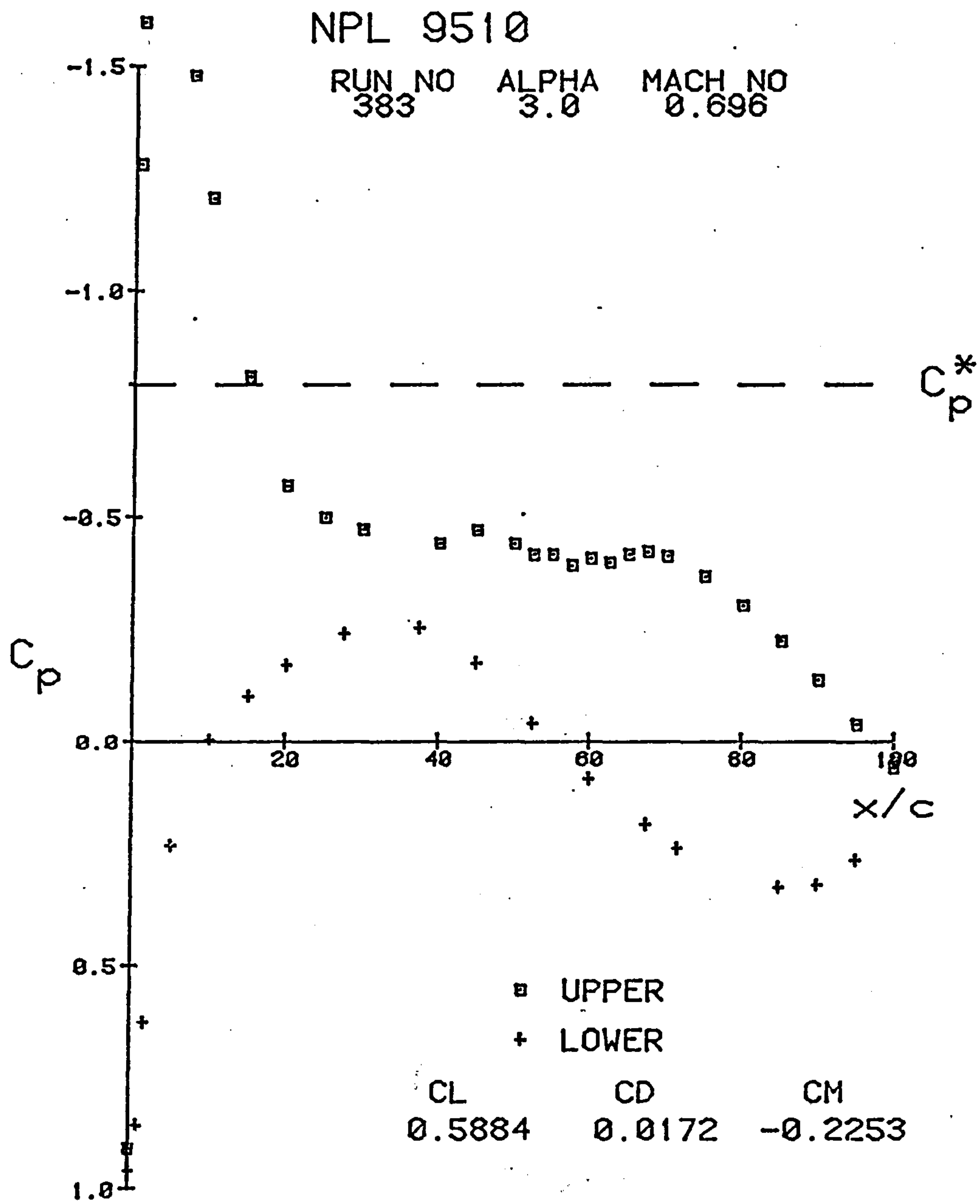

NPL 9510
RUN NO 384
ALPHA 4.0
MACH NO 0.696
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.7026
CD 0.0292
CM -0.2491

Fig. 3.34.



NPL 9510
RUN NO 383
ALPHA 3.0
MACH NO 0.696
-1.5
-1.0
-0.5
0.0
0.5
1.0
20
40
60
80
100
Cp*
Cp
x/c
UPPER
LOWER
CL 0.5884
CD 0.0172
CM -0.2253

Fig. 3.35.

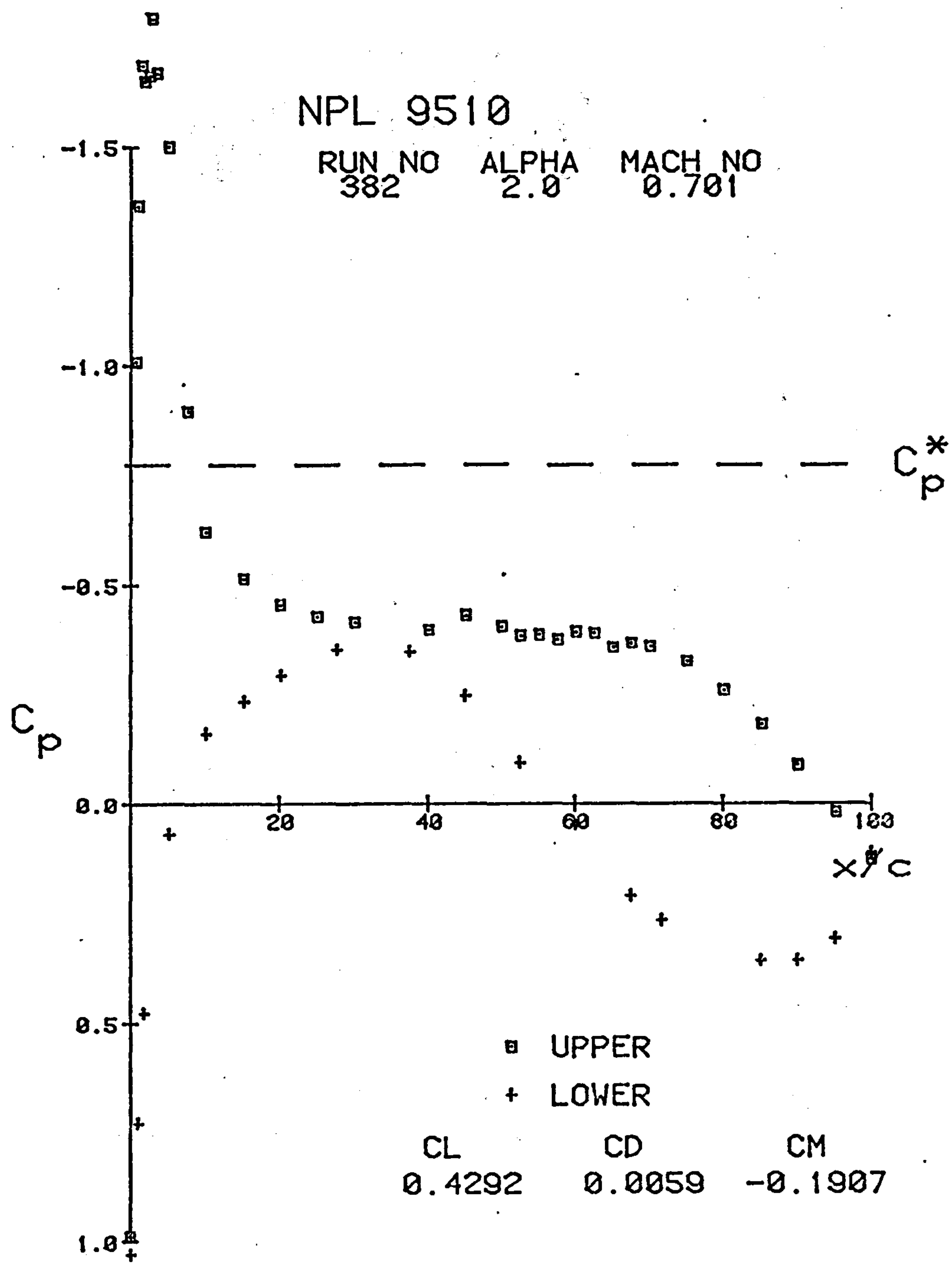

NPL 9510
RUN NO 382
ALPHA 2.0
MACH NO 0.701
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4292
CD 0.0059
CM -0.1907

Fig. 3.36.

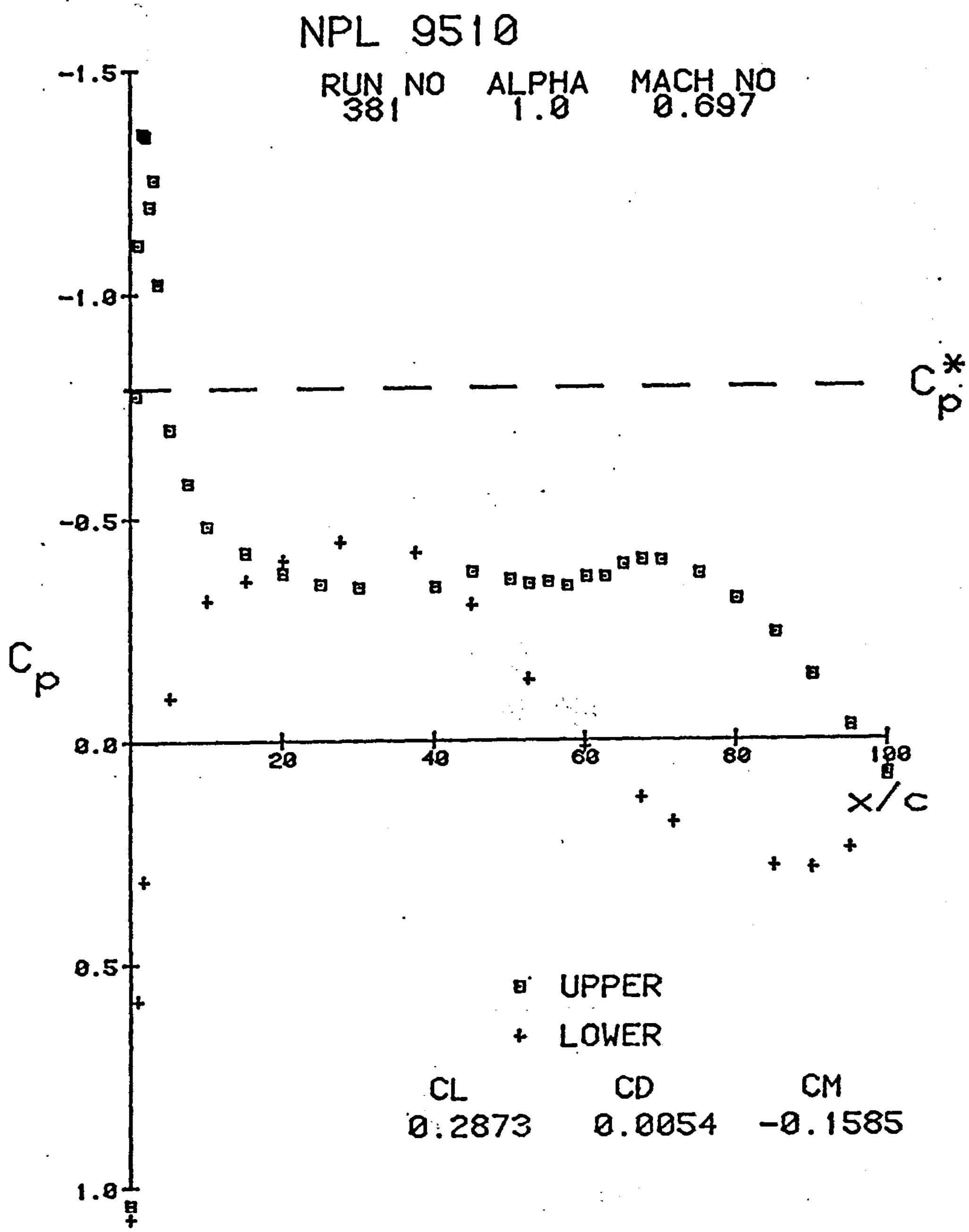

NPL 9510
RUN NO ALPHA MACH NO
381 1.0 0.697
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL CD CM
0.2873 0.0054 -0.1585

Fig. 3.37.

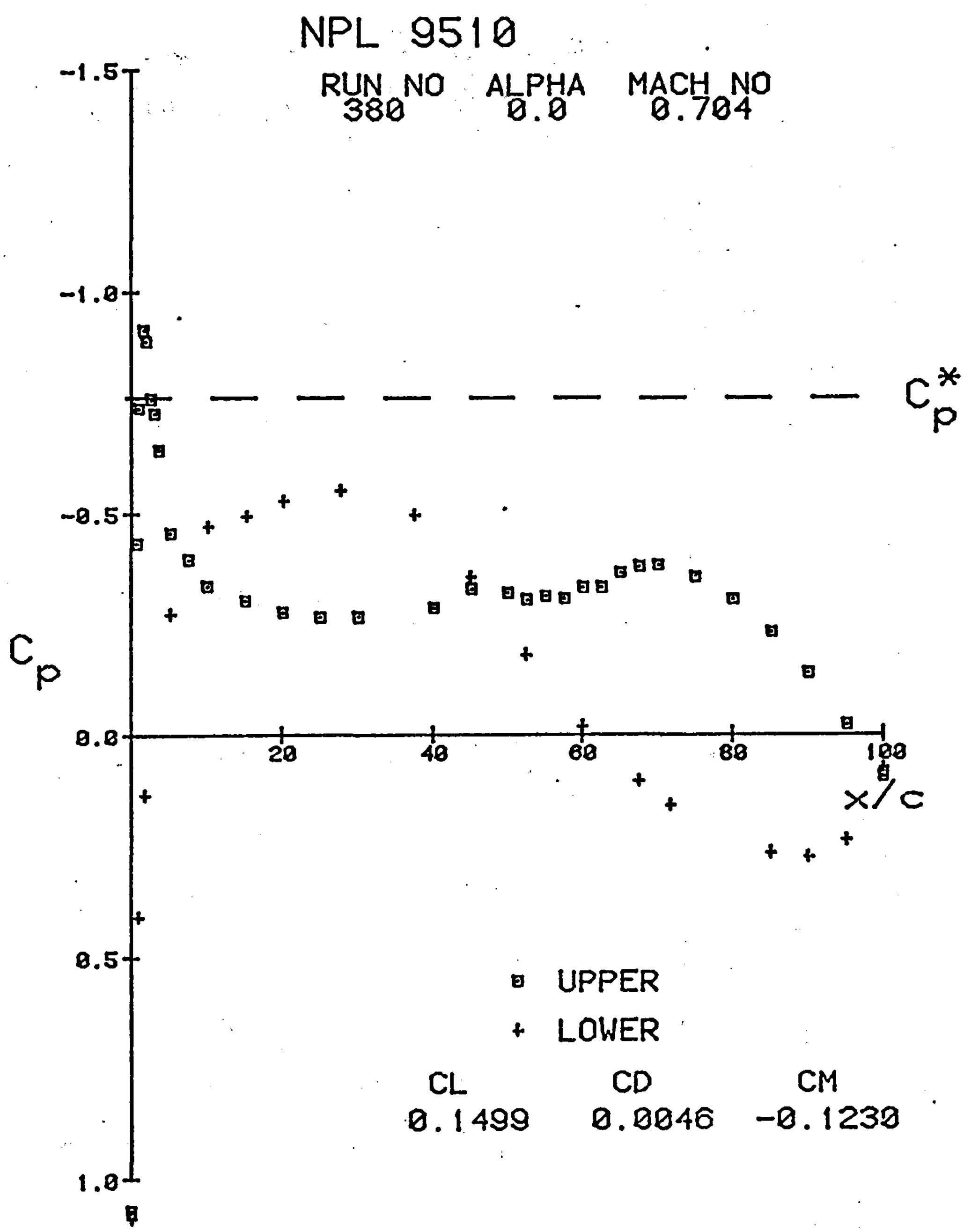
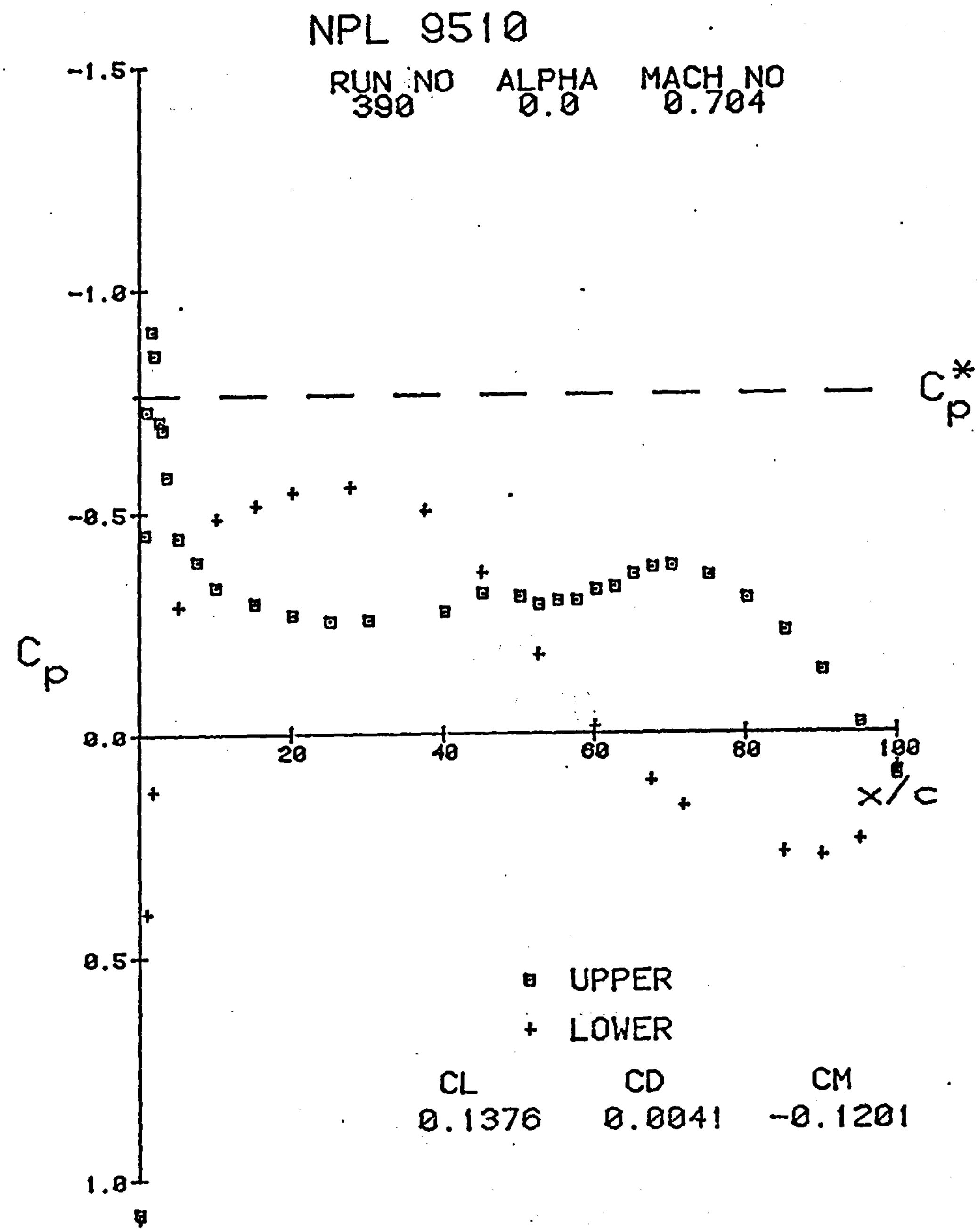

NPL 9510
RUN NO 380
ALPHA 0.0
MACH NO 0.704
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1499
CD 0.0046
CM -0.1230

Fig. 3.38.



NPL 9510
RUN NO 390
ALPHA 0.0
MACH NO 0.704
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
Cp*
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1376
CD 0.0041
CM -0.1201

Fig. 3.39.

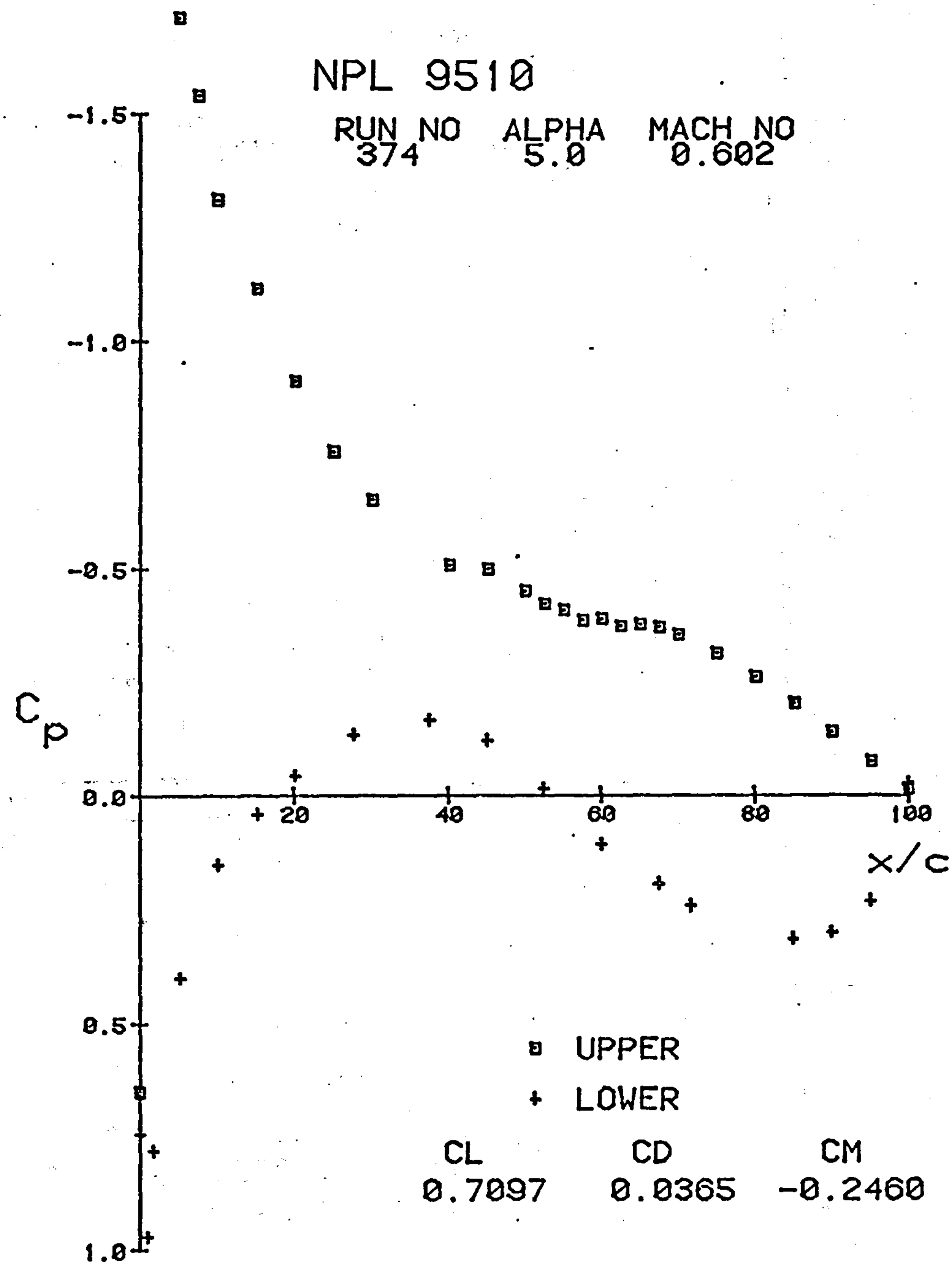

NPL 9510
RUN NO 374
ALPHA 5.0
MACH NO 0.602
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.7097
CD 0.0365
CM -0.2460

Fig. 3.40.

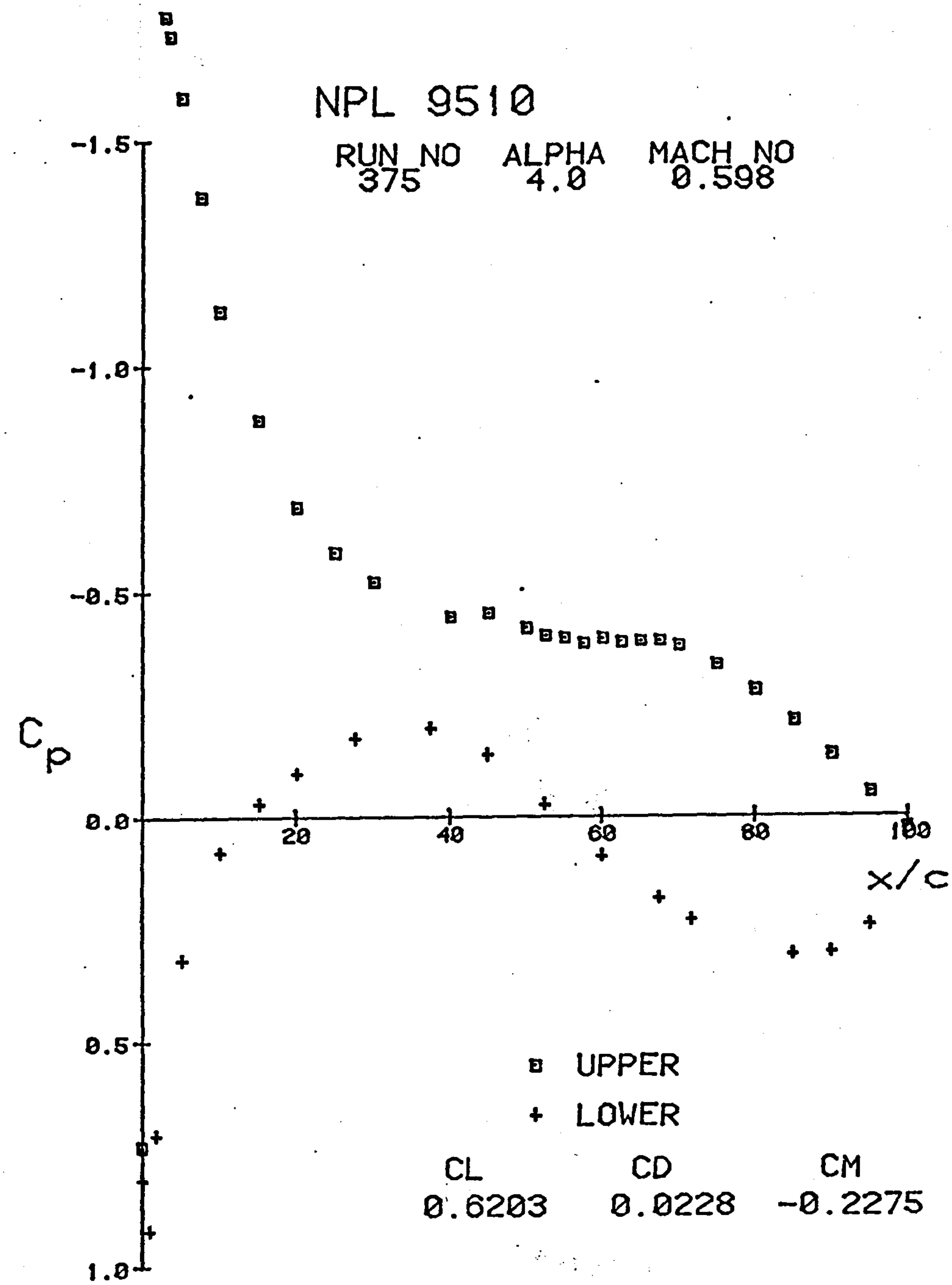

NPL 9510
RUN NO ALPHA MACH NO
375 4.0 0.598
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL CD CM
0.6203 0.0228 -0.2275

Fig. 3.41.

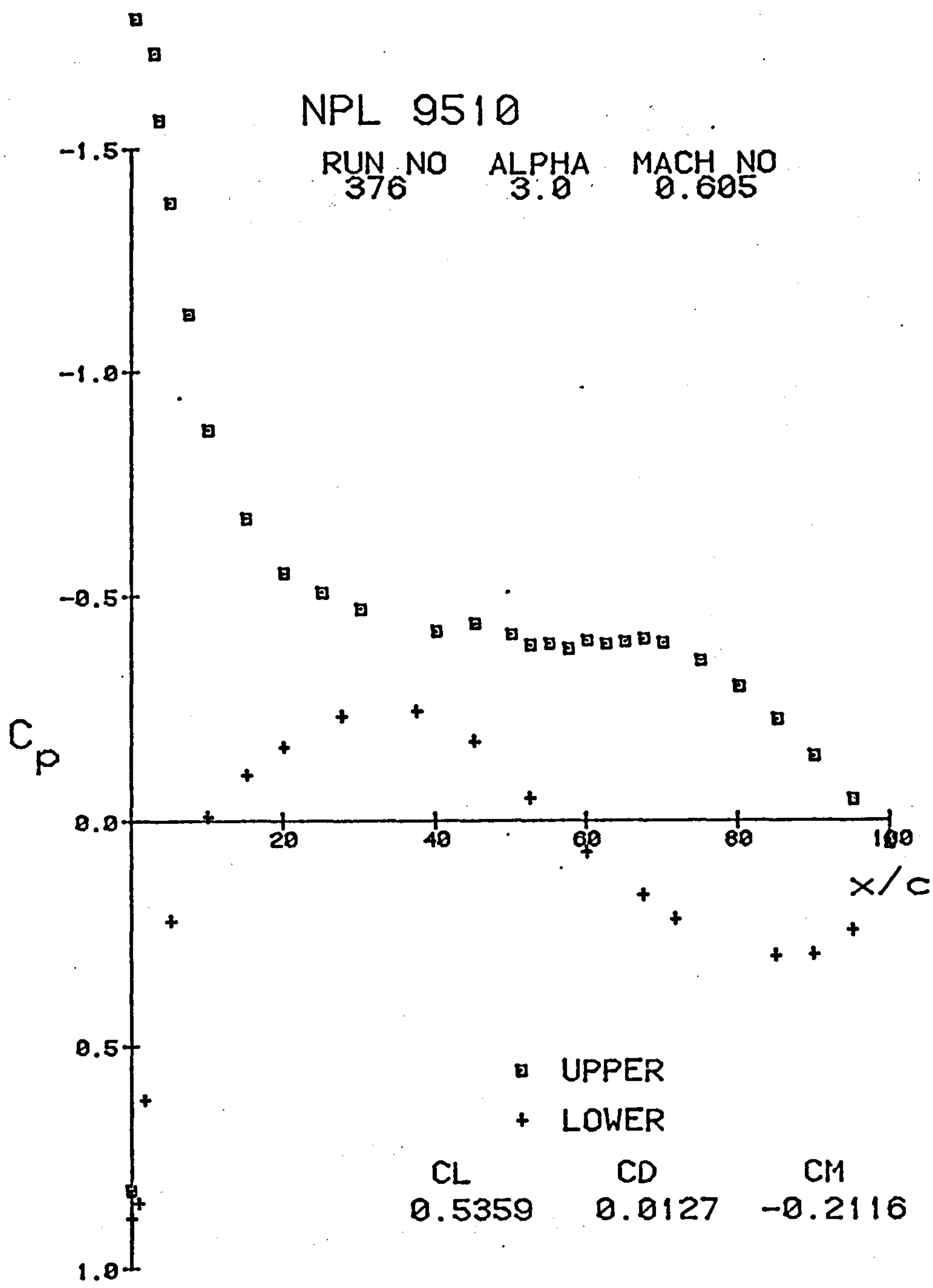
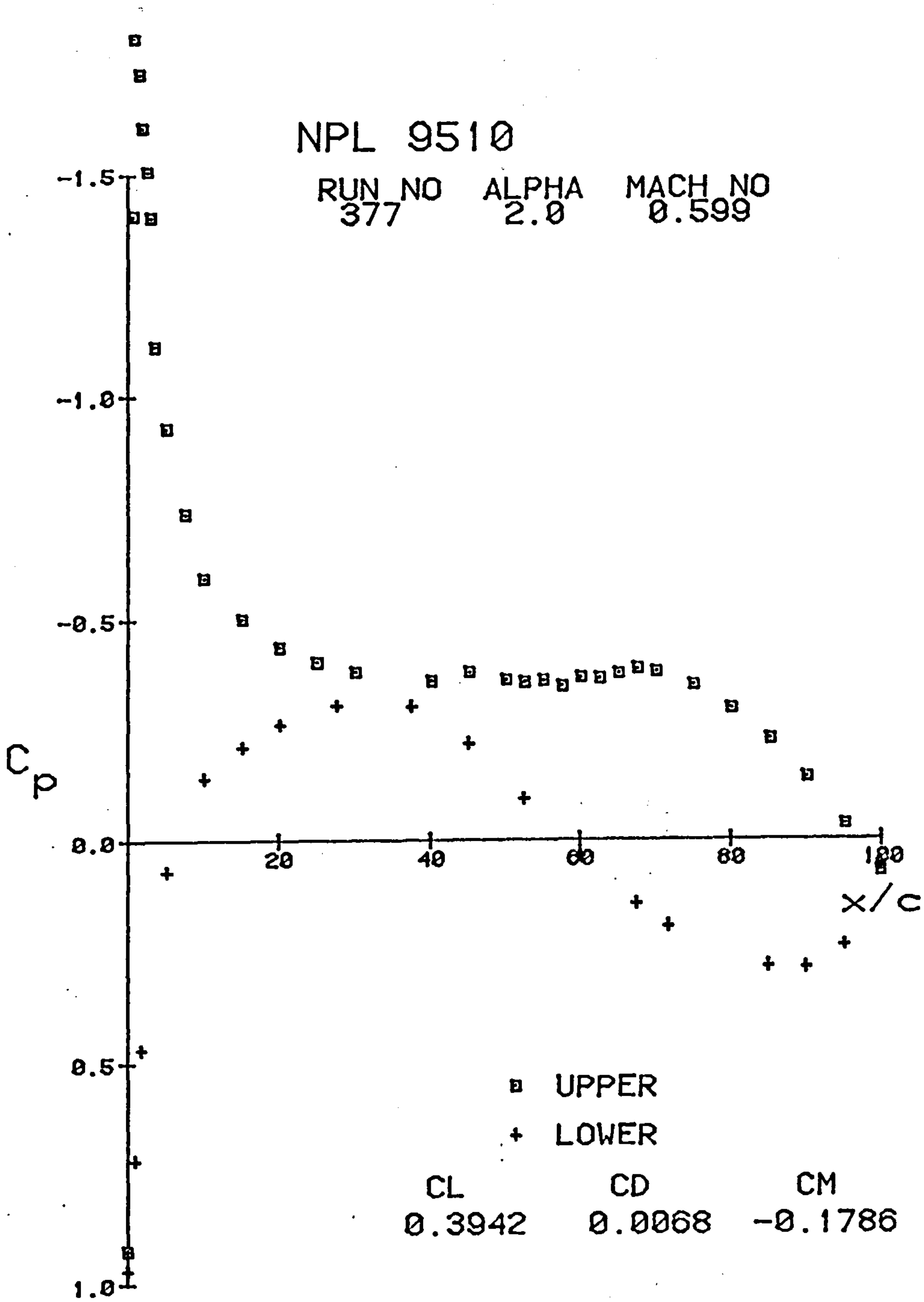

NPL 9510
RUN NO 376
ALPHA 3.0
MACH NO 0.605
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.5359
CD 0.0127
CM -0.2116

Fig. 3.42.



NPL 9510
RUN NO 377
ALPHA 2.0
MACH NO 0.599
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.3942
CD 0.0068
CM -0.1786

Fig. 3.43.

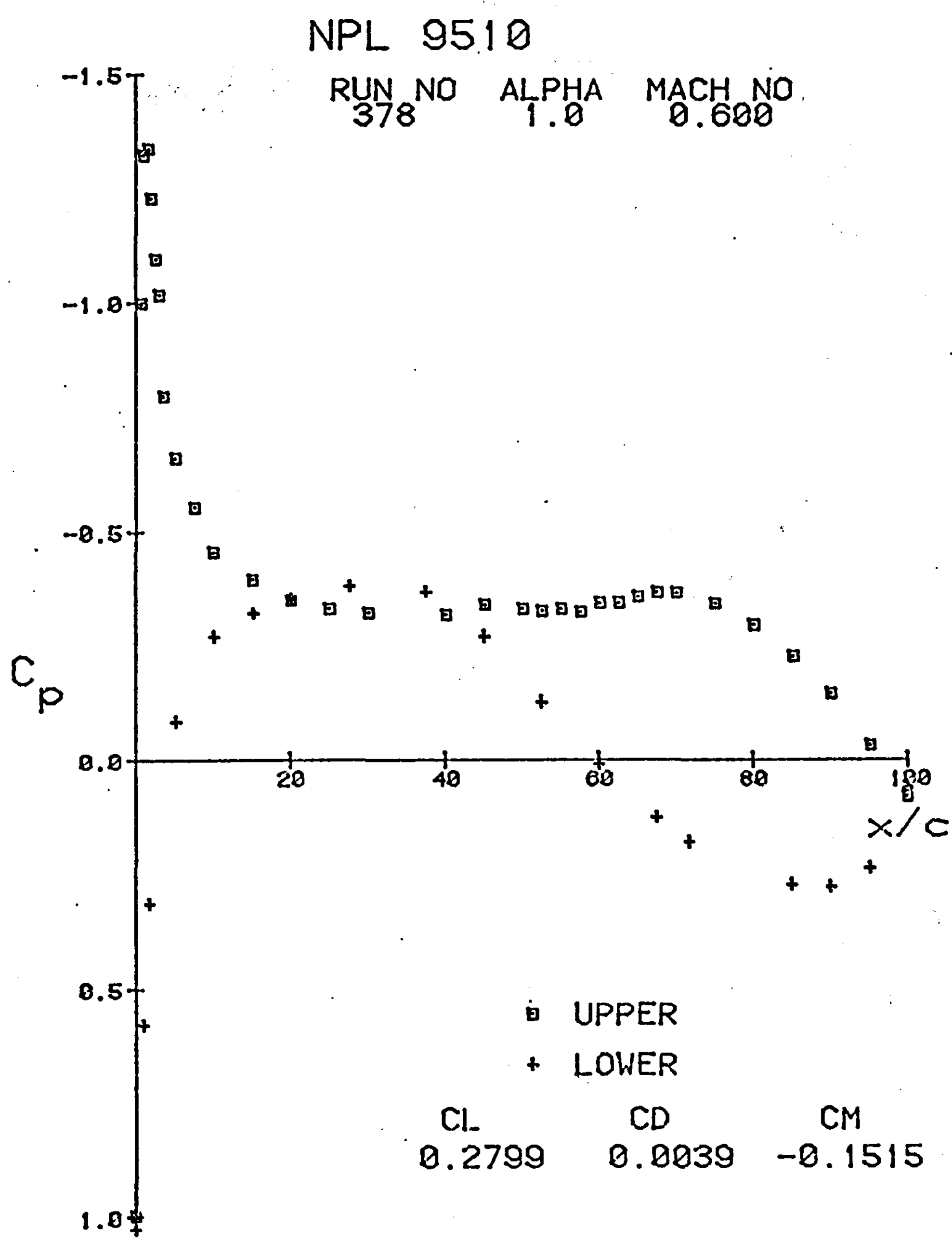
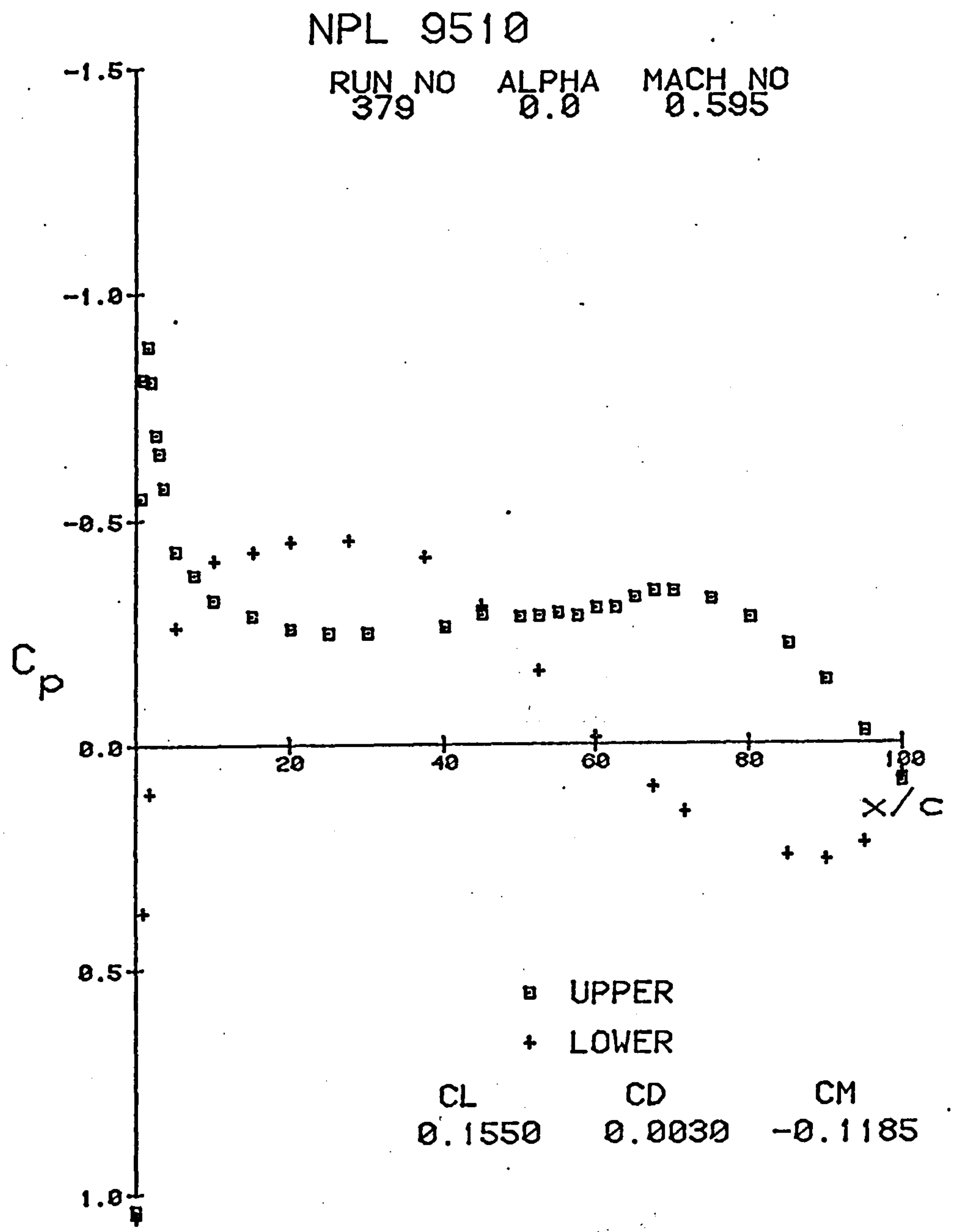

NPL 9510
RUN NO 378
ALPHA 1.0
MACH NO 0.600
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.2799
CD 0.0039
CM -0.1515

Fig. 3.44.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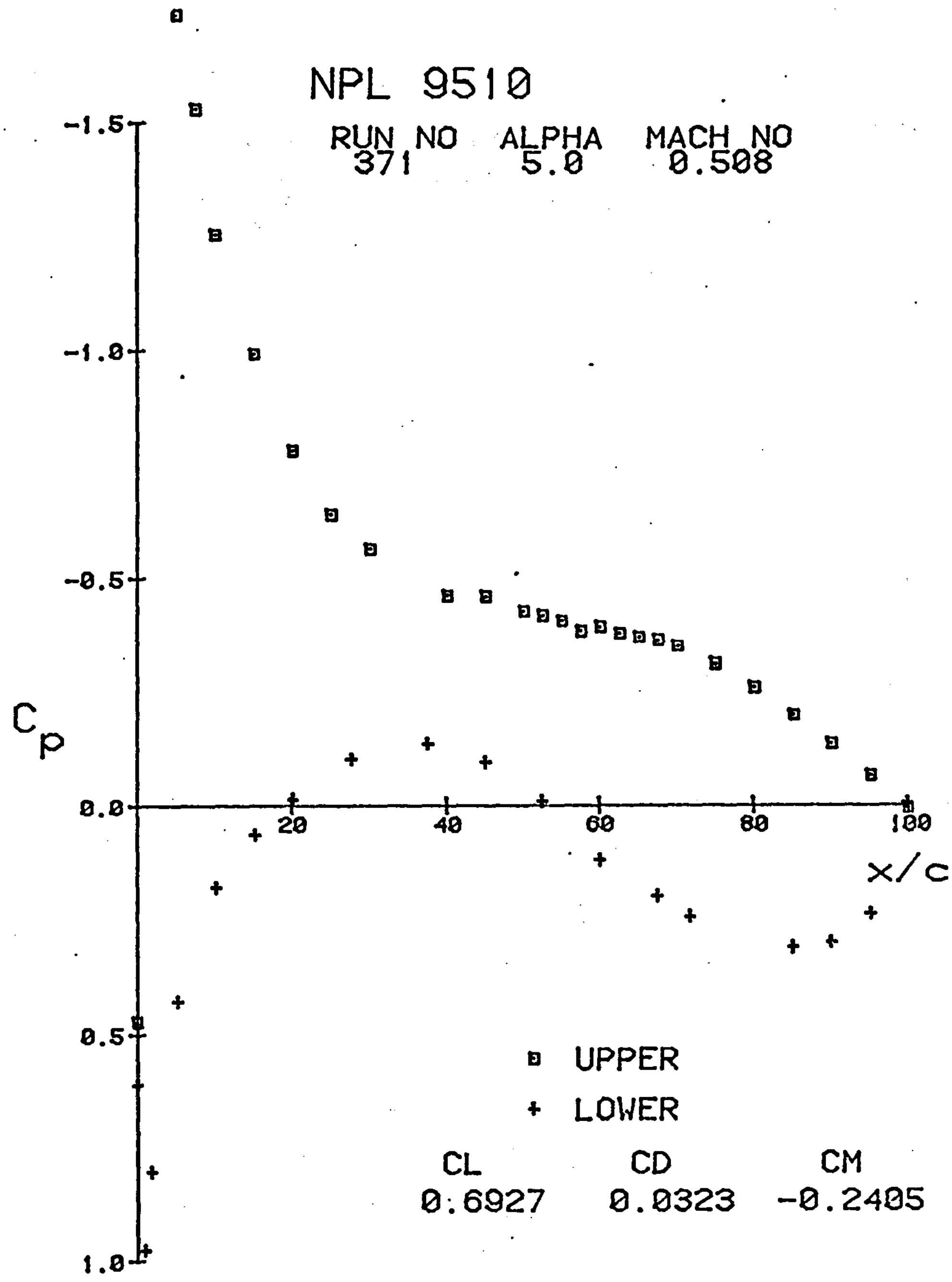
RUN NO 379
ALPHA 0.0
MACH NO 0.595
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1550
CD 0.0030
CM -0.1185

Fig. 3.45.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RUN NO 371
ALPHA 5.0
MACH NO 0.508
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6927
CD 0.0323
CM -0.2405

Fig. 3.46.

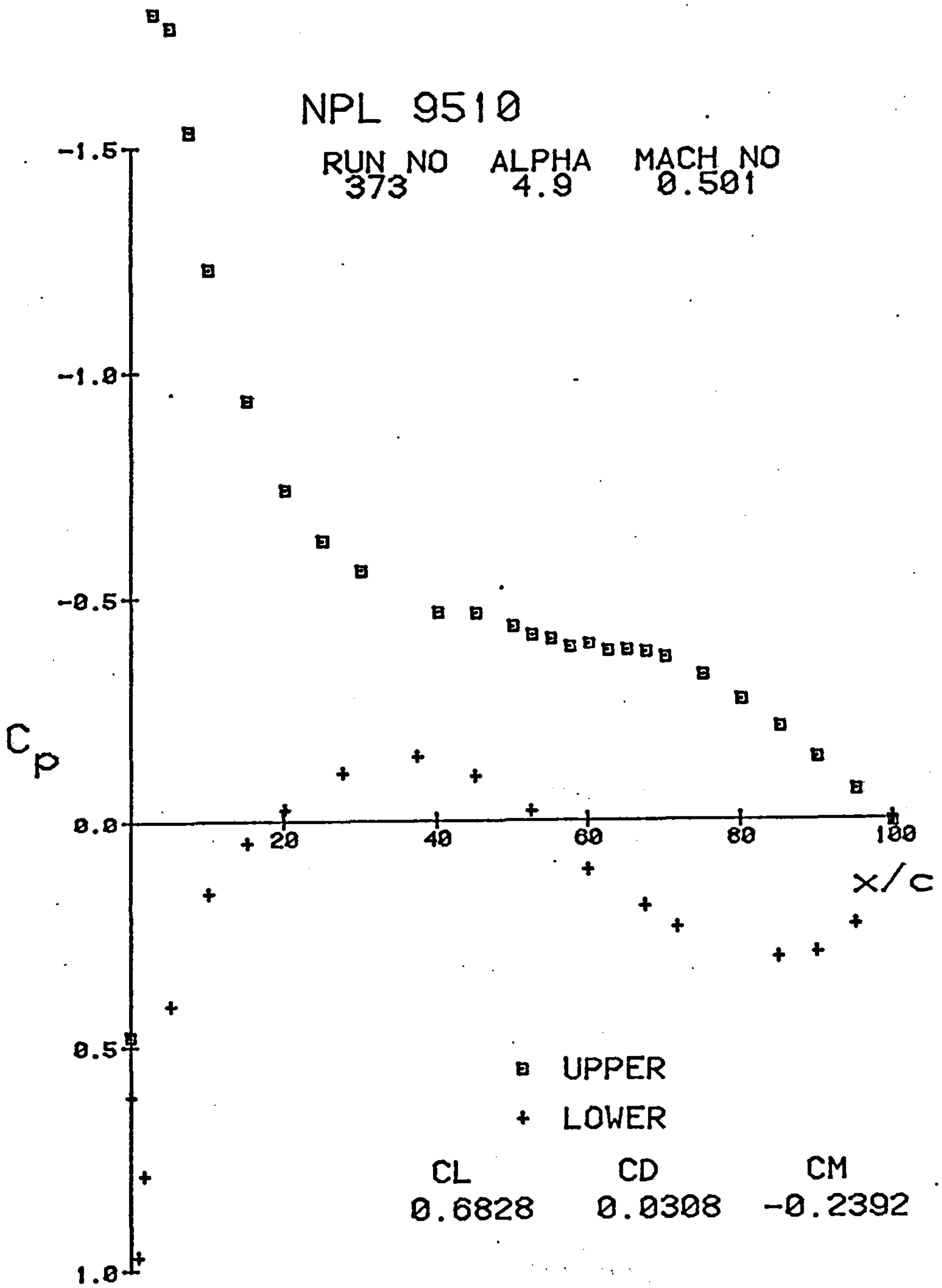

NPL 9510
RUN NO 373
ALPHA 4.9
MACH NO 0.501
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6828
CD 0.0308
CM -0.2392

Fig. 3.47.

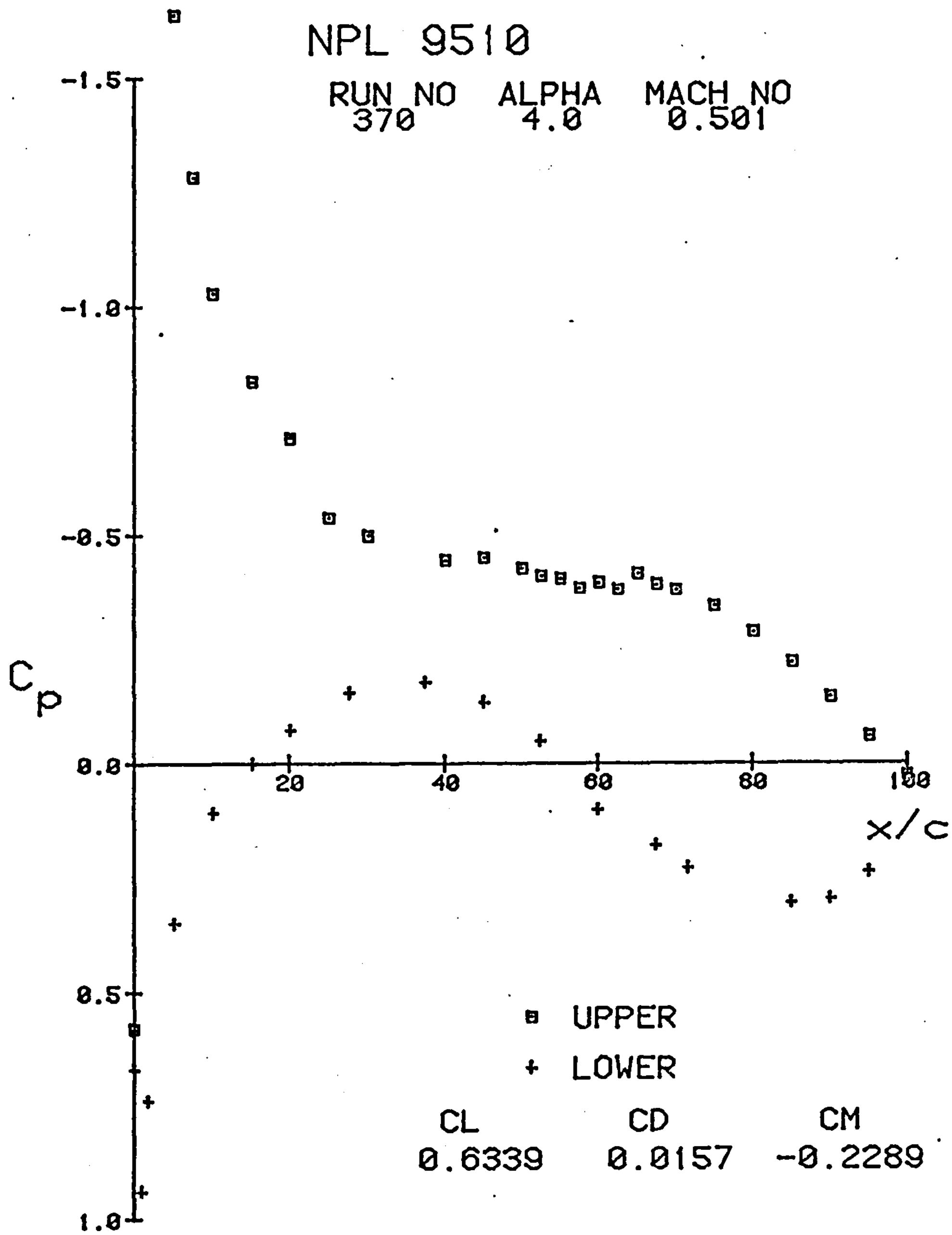

NPL 9510
RUN NO 370
ALPHA 4.0
MACH NO 0.501
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.6339
CD 0.0157
CM -0.2289

Fig. 3.48.

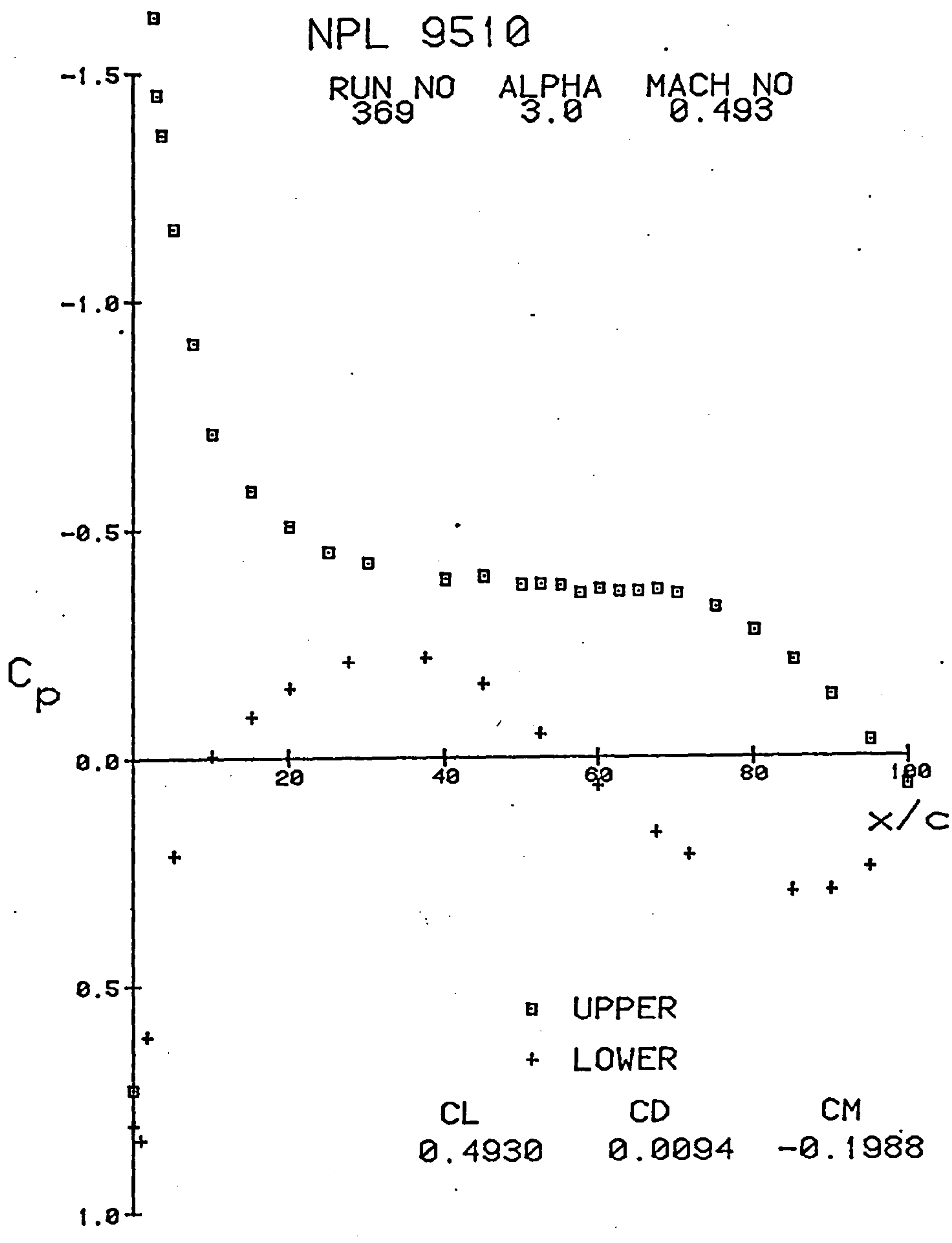
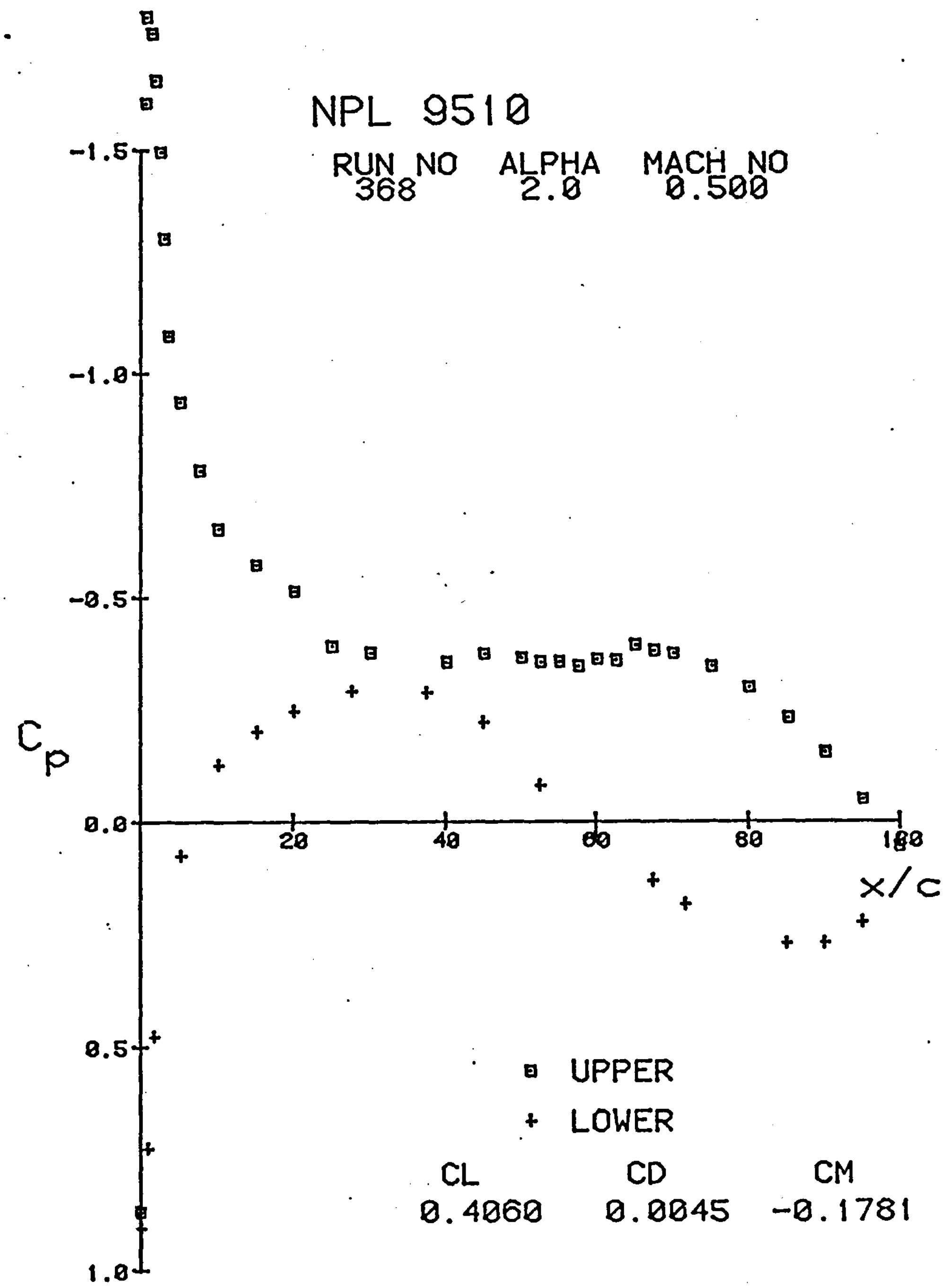

NPL 9510
RUN NO 369
ALPHA 3.0
MACH NO 0.493
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4930
CD 0.0094
CM -0.1988

Fig. 3.49.



NPL 9510
RUN NO 368
ALPHA 2.0
MACH NO 0.500
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.4060
CD 0.0045
CM -0.1781

Fig. 3.50.

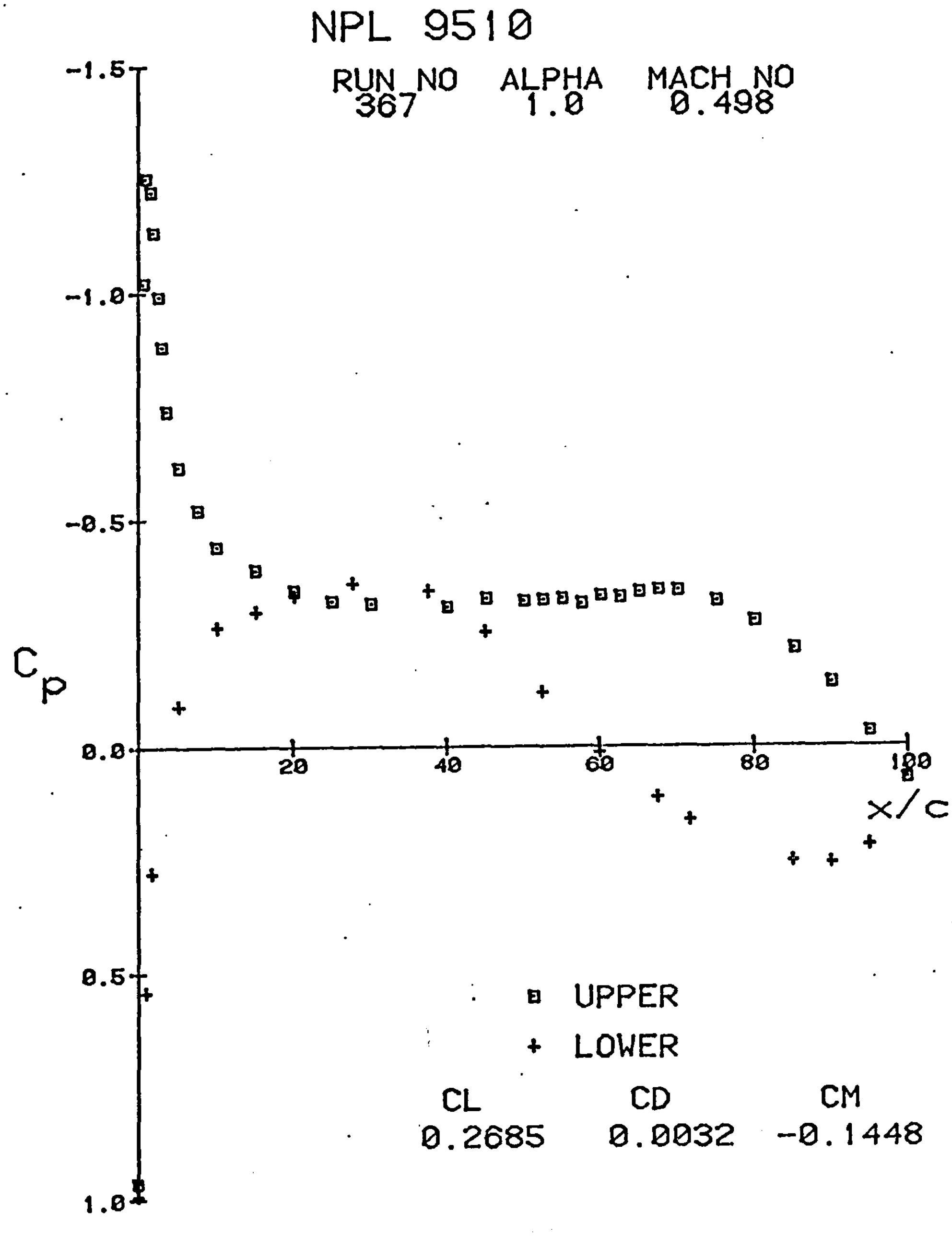
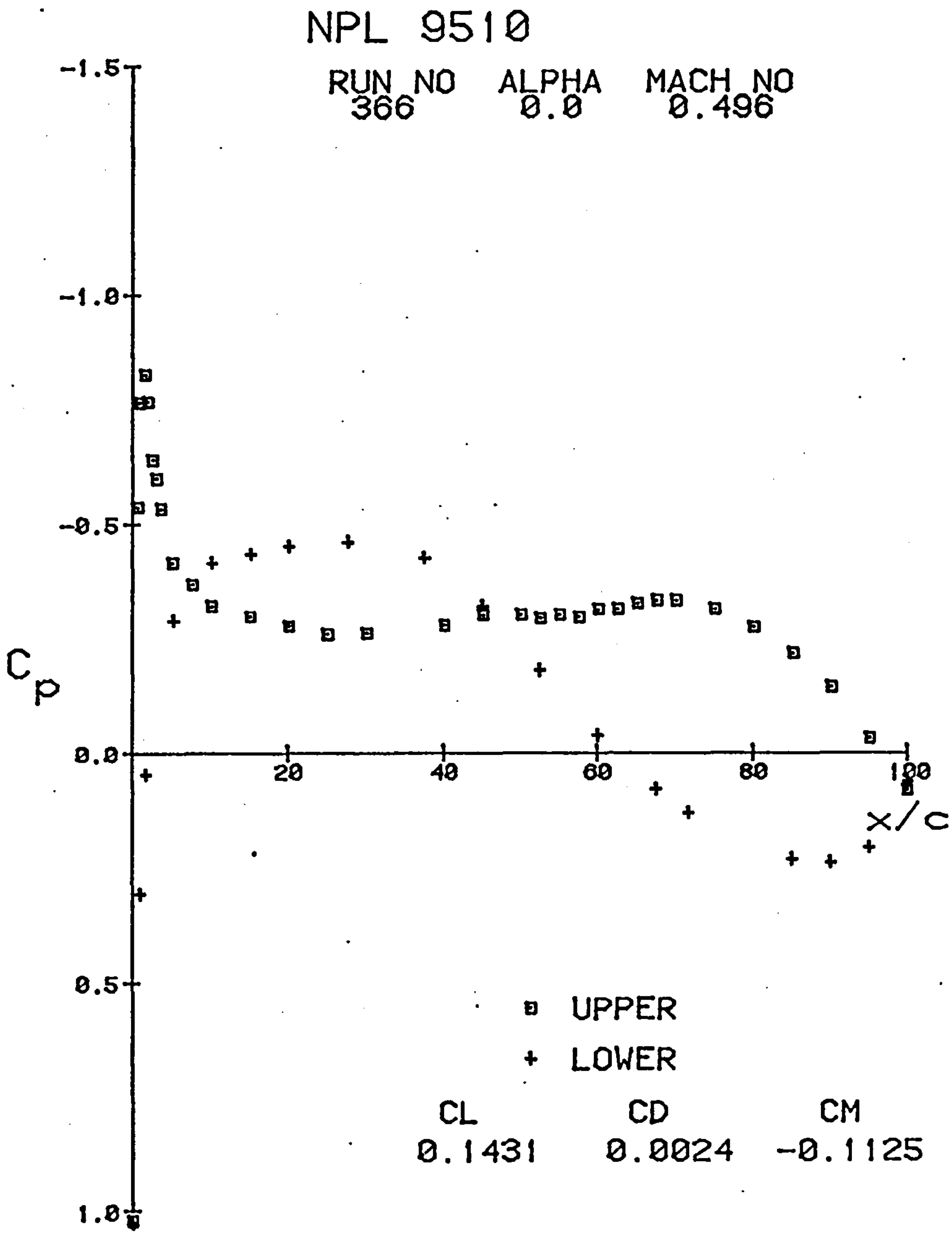

NPL 9510
RUN NO 367
ALPHA 1.0
MACH NO 0.498
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.2685
CD 0.0032
CM -0.1448

Fig. 3.51.



NPL 9510
RUN NO 366
ALPHA 0.0
MACH NO 0.496
-1.5
-1.0
-0.5
0.0
0.5
1.0
Cp
20
40
60
80
100
x/c
UPPER
LOWER
CL 0.1431
CD 0.0024
CM -0.1125

Fig. 3.52.

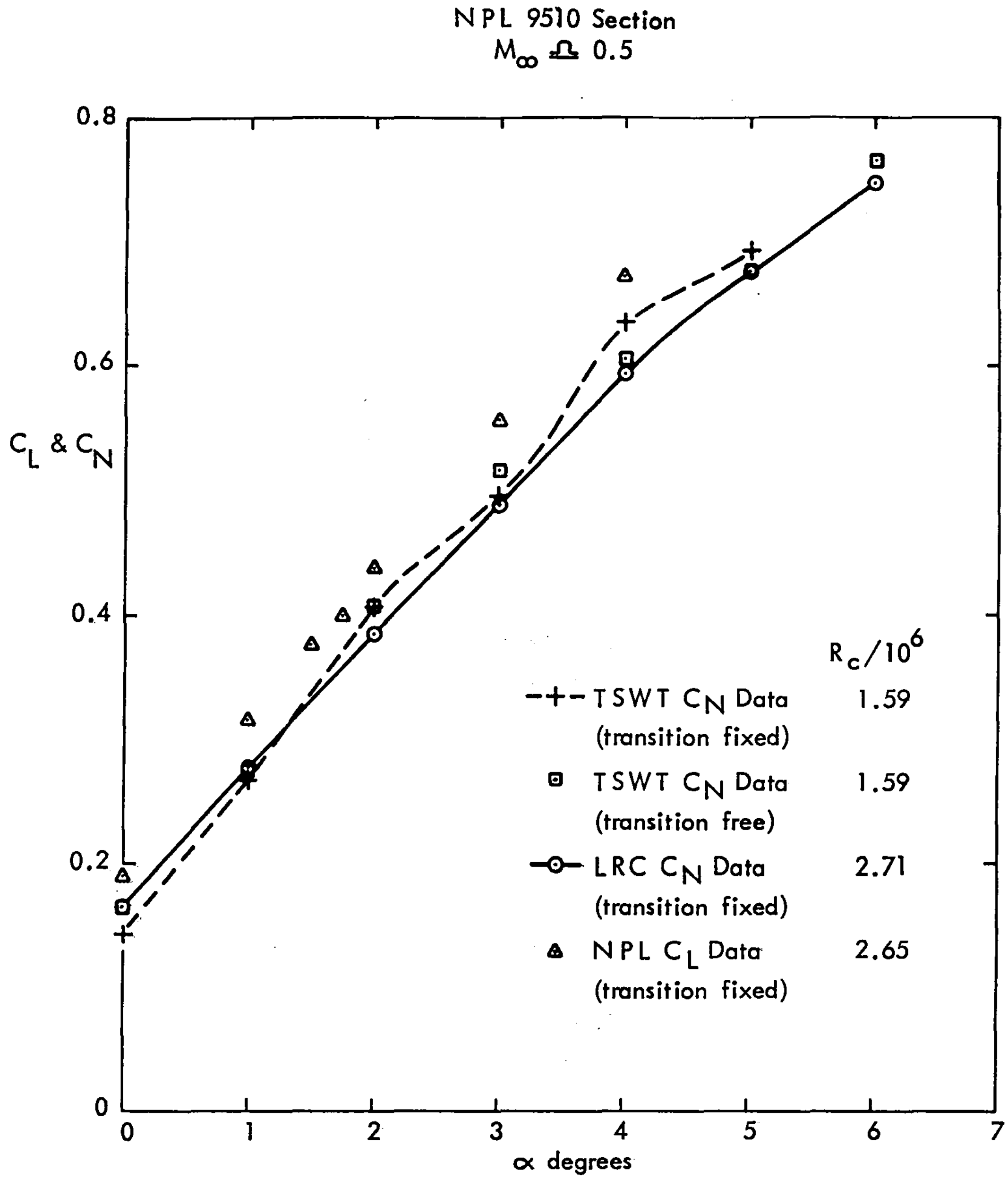


FIG. 4a VARIATIONS OF C_L AND C_N WITH ANGLE OF ATTACK; $M_\infty \approx 0.5$

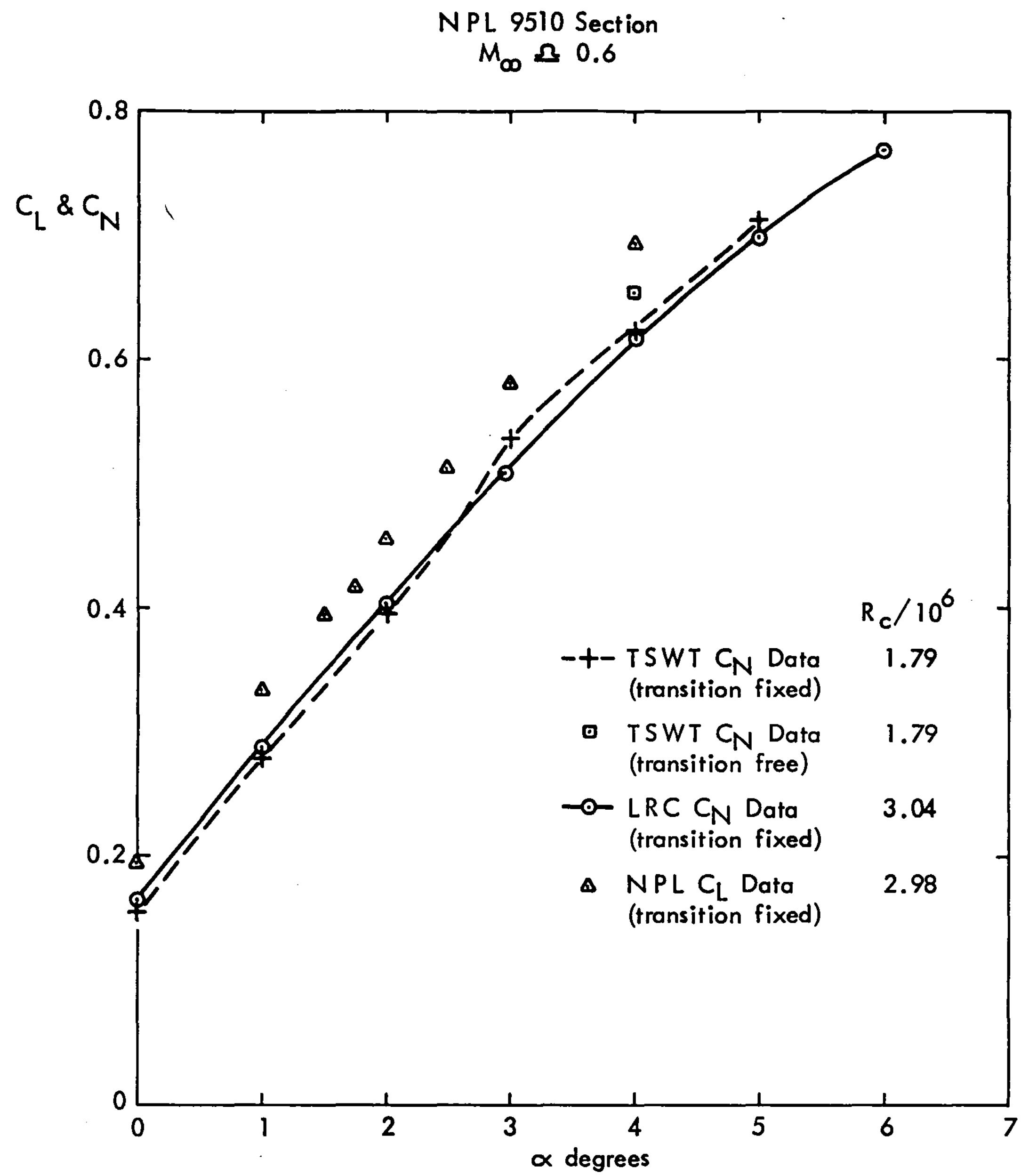


FIG. 4b VARIATIONS OF C_L AND C_N WITH ANGLE OF ATTACK; $M_\infty \simeq 0.6$

N PL 9510 Section
$M_\infty \simeq 0.7$

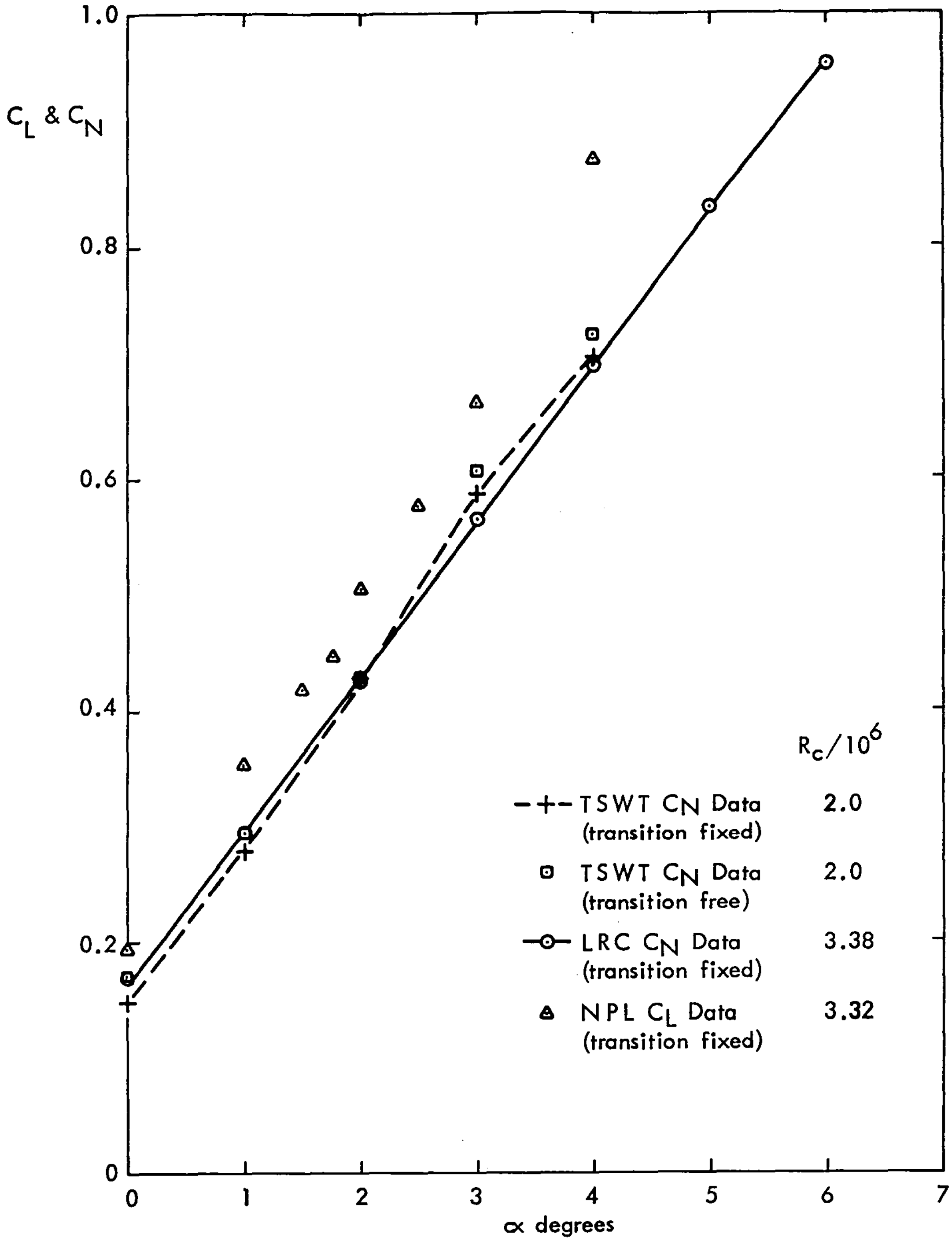


FIG. 4c VARIATIONS OF C_L AND C_N WITH ANGLE OF ATTACK; $M_\infty \simeq 0.7$

NPL 9510 Section
$M_\infty \cong 0.75$

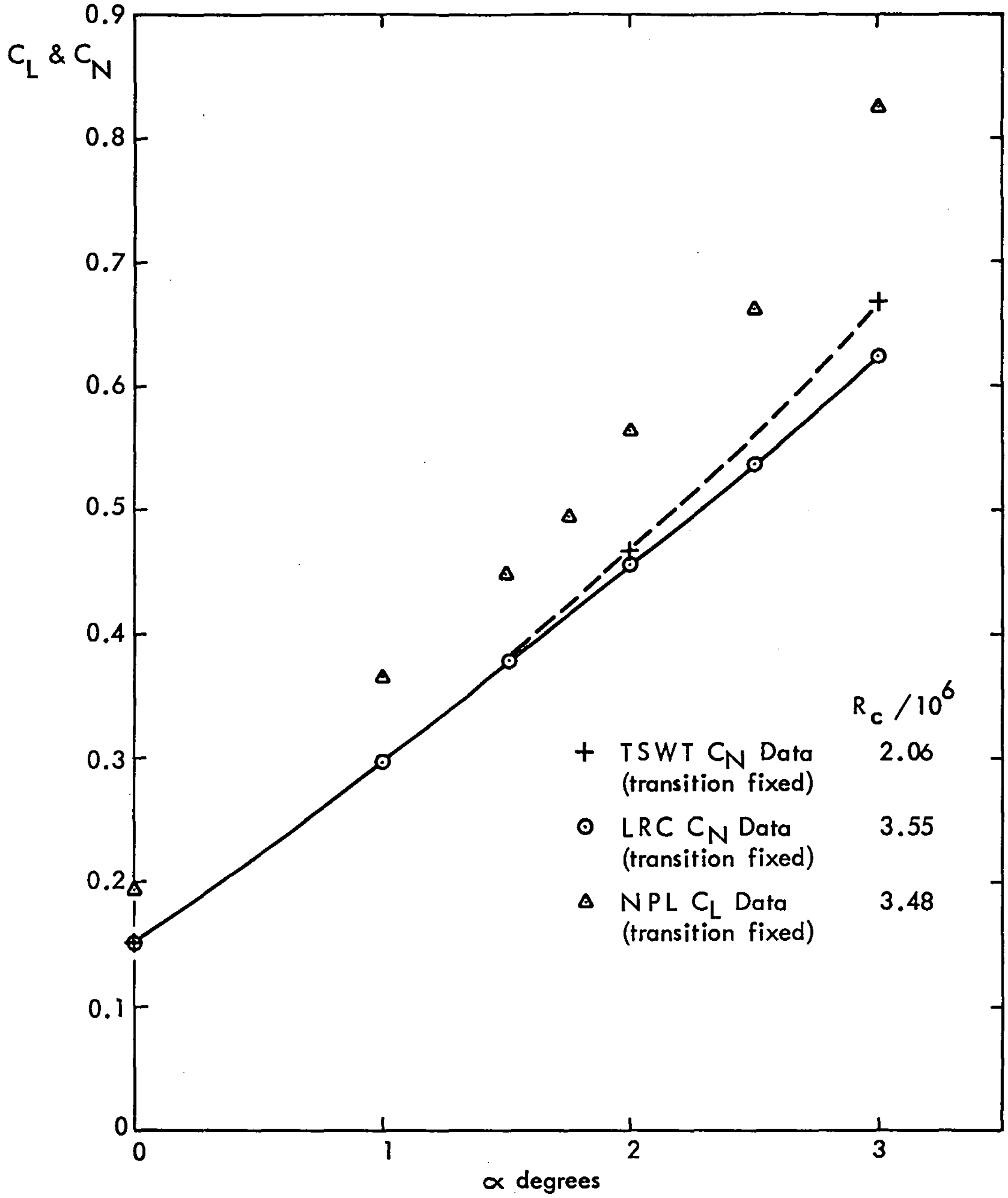


FIG. 4d VARIATIONS OF C_L AND C_N WITH ANGLE OF ATTACK; $M_\infty \cong 0.75$

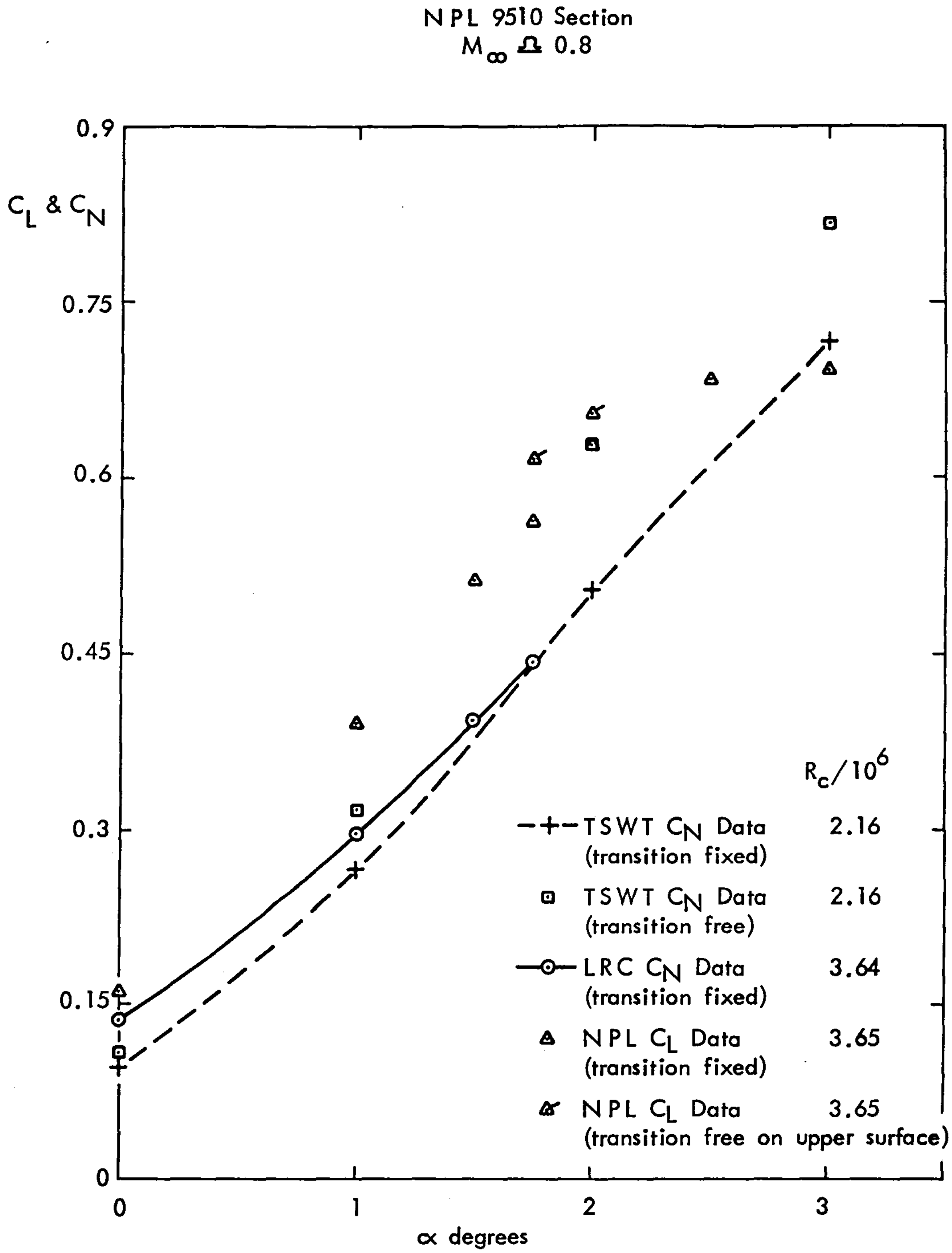


FIG. 4e VARIATIONS OF C_L AND C_N WITH ANGLE OF ATTACK; $M_\infty \simeq 0.8$

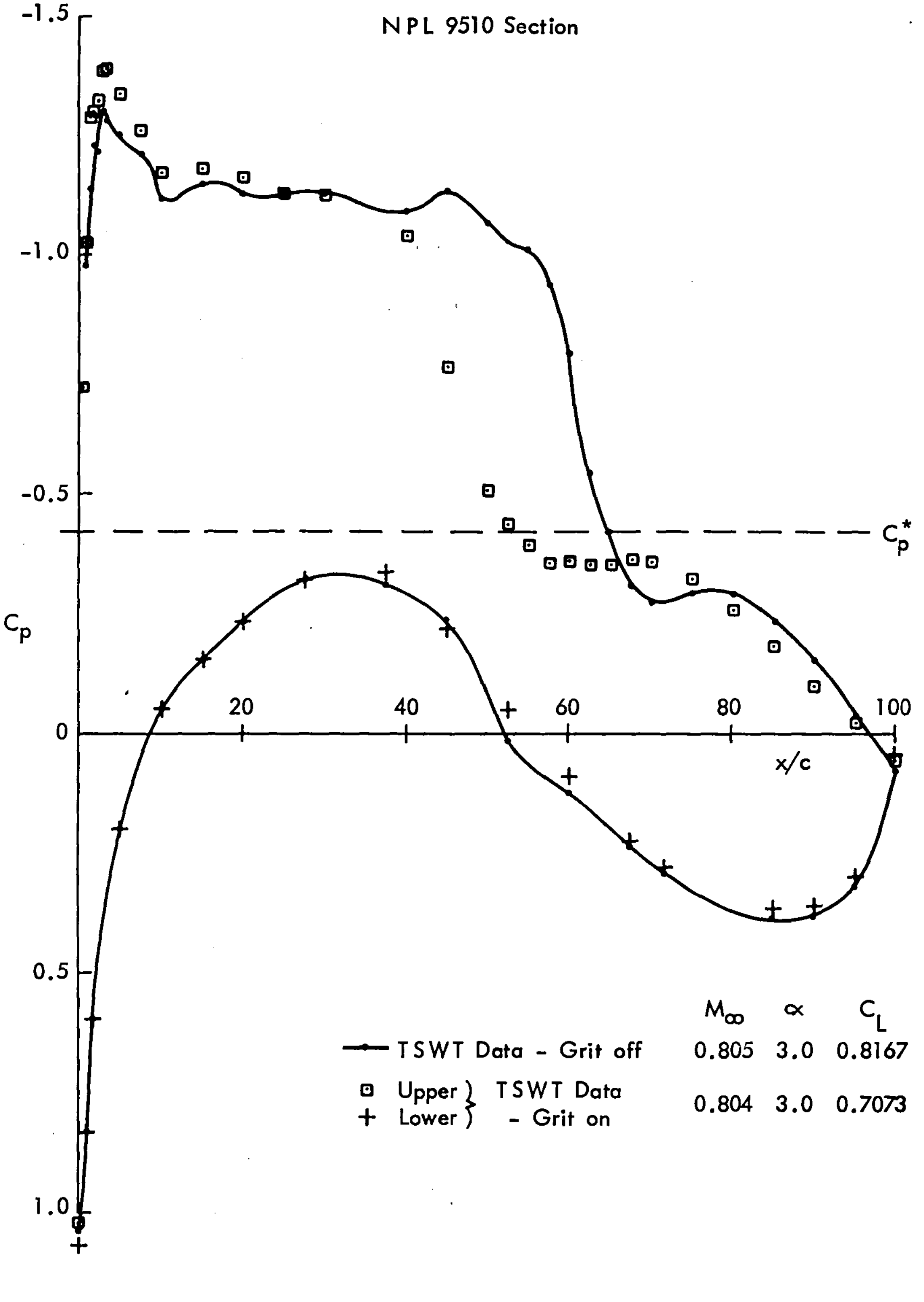


FIG. 5 COMPARISON OF MODEL PRESSURE DISTRIBUTIONS WITH TRANSITION FIXED AND TRANSITION FREE : $M_\infty \simeq 0.8$; $\alpha = 3^\circ$

N P L 9510 Section

$\alpha = 2^{o}$

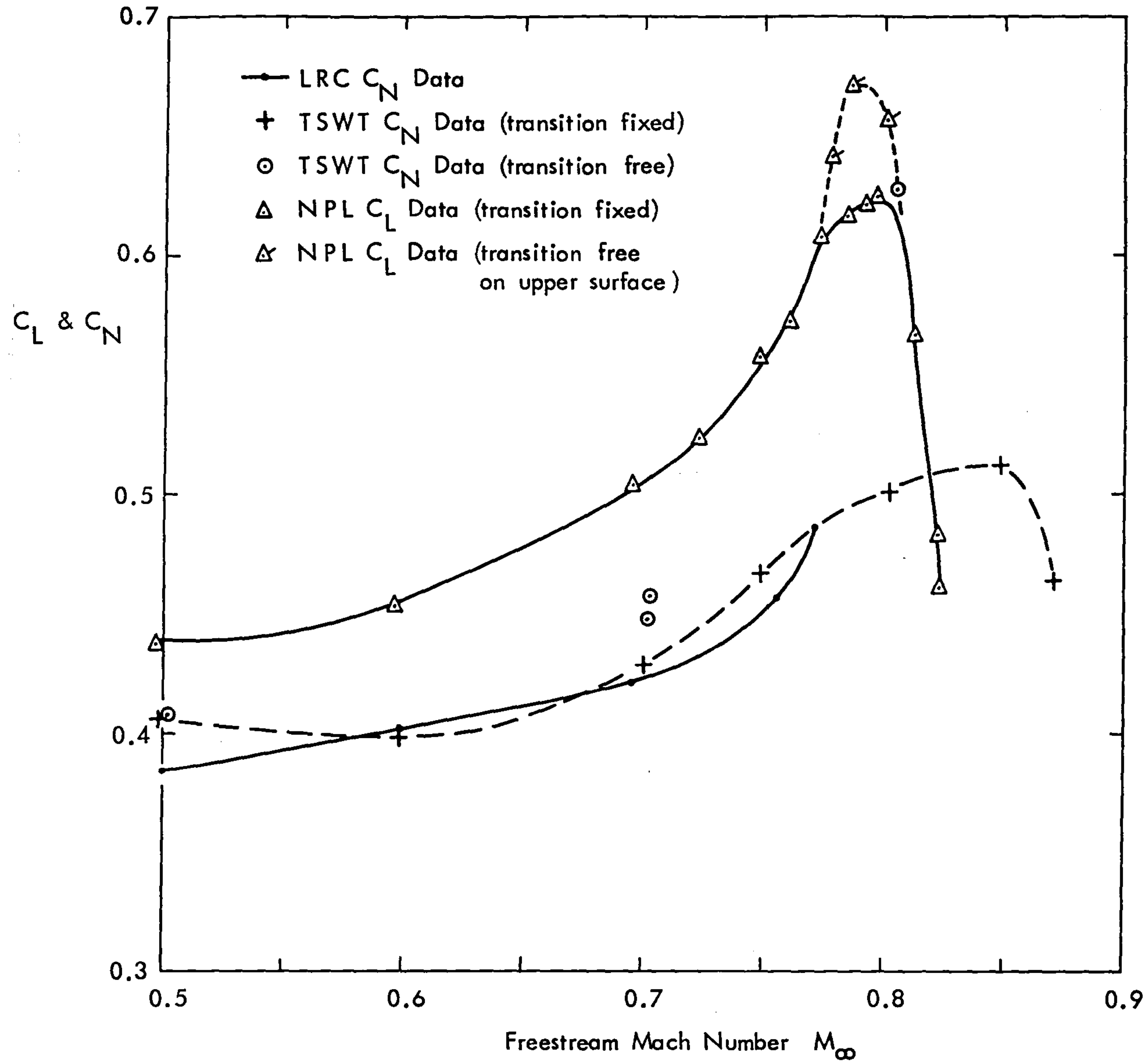


FIG. 6 VARIATION OF NORMAL FORCE WITH MACH NUMBER; $\alpha = 2^{o}$

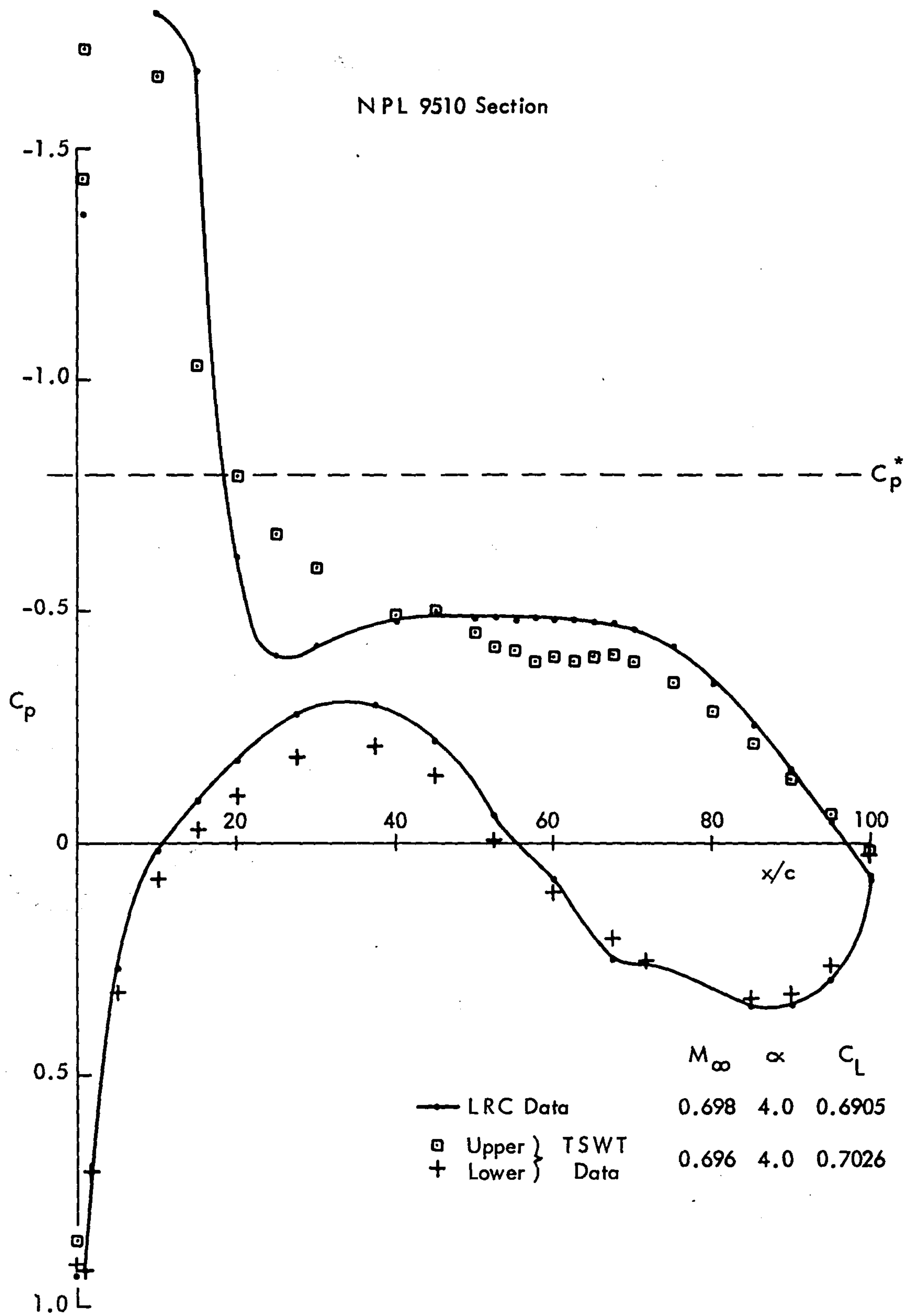


FIG. 7 COMPARISON OF TSWT AND LRC MODEL PRESSURE DISTRIBUTIONS ; $M_\infty \simeq 0.7$; $\alpha \simeq 4°$

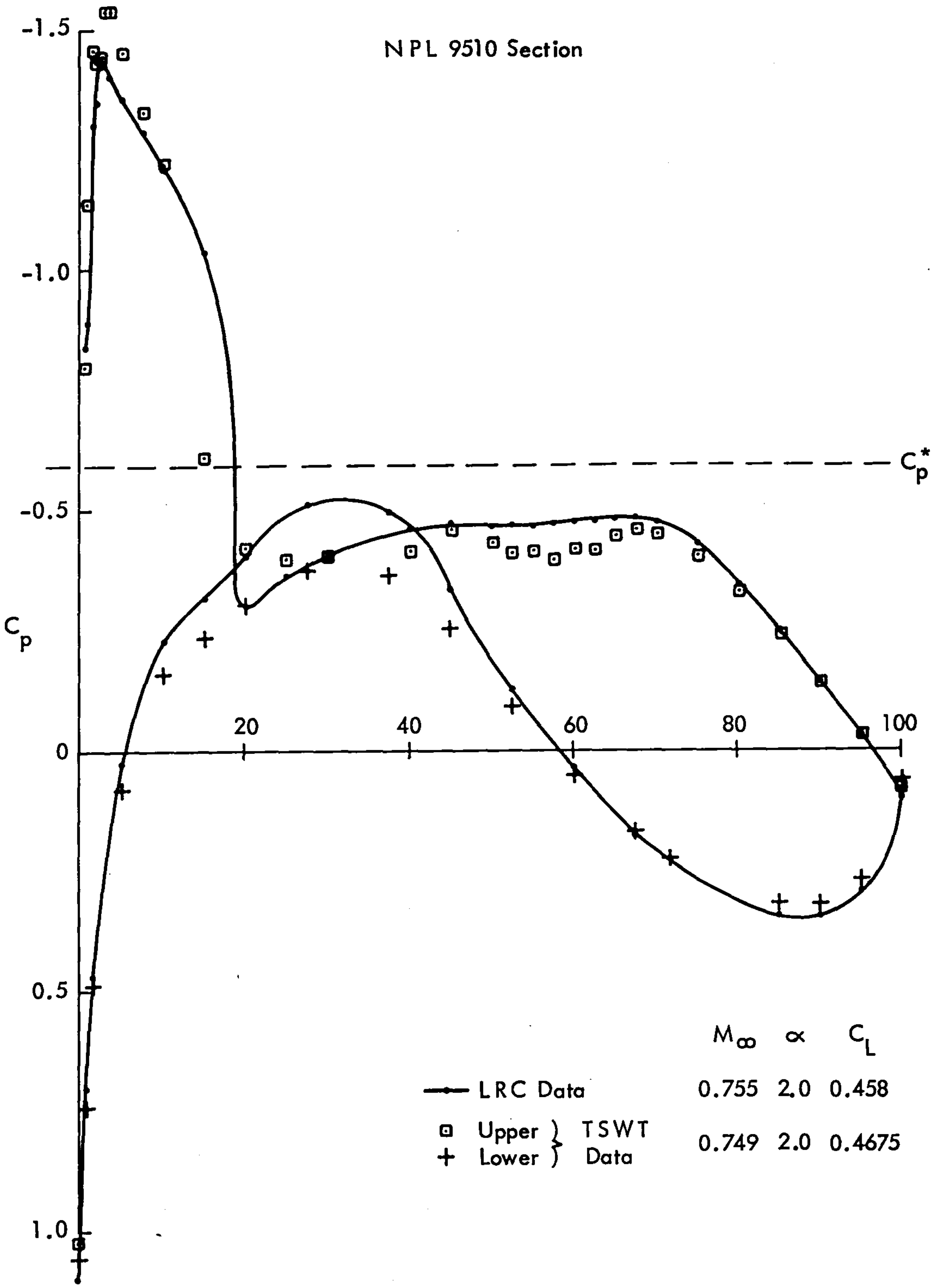


FIG. 8 COMPARISON OF TSWT AND LRC MODEL PRESSURE DISTRIBUTIONS : $M_\infty \simeq 0.75$; $\alpha \simeq 2°$

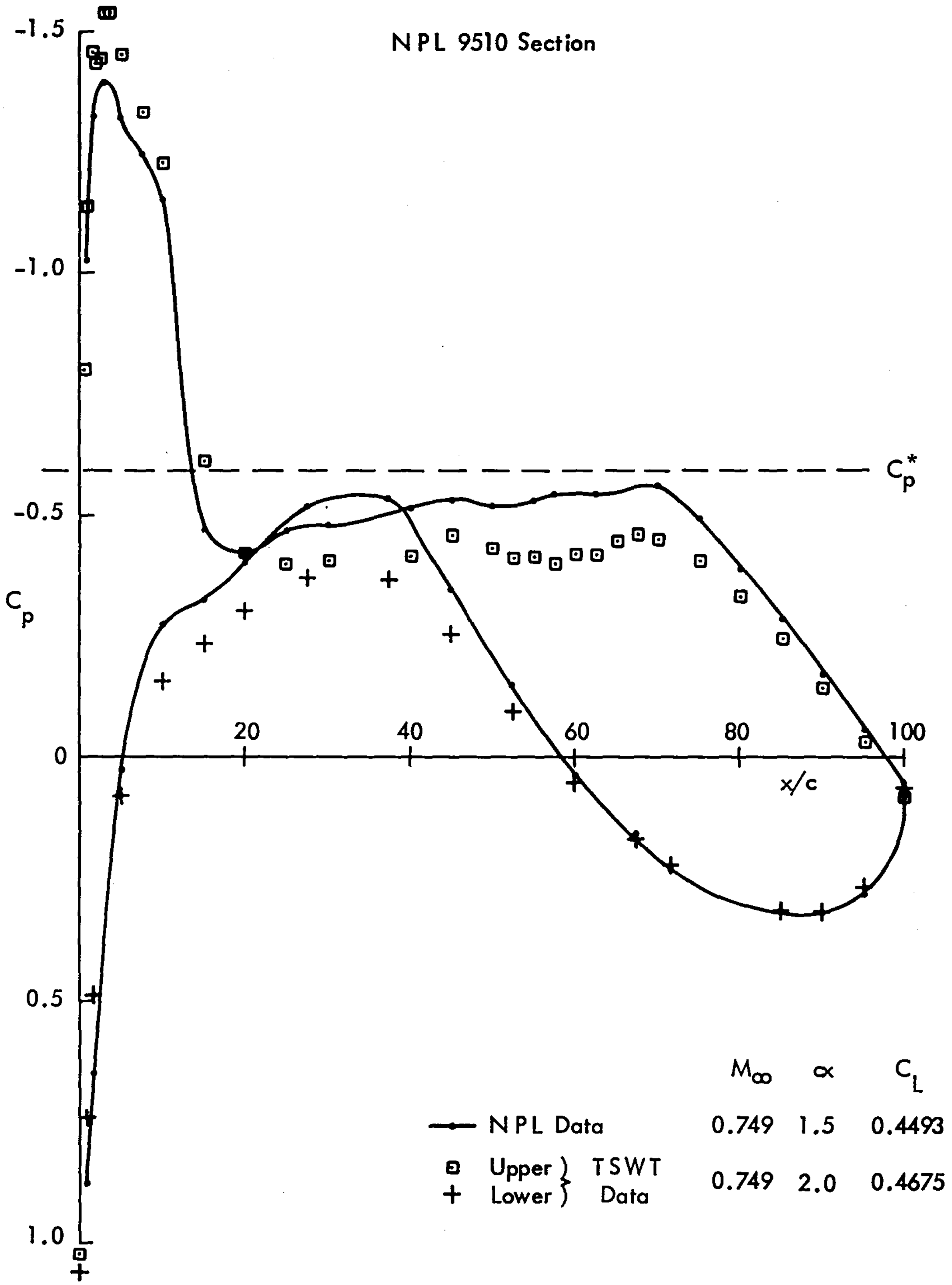


FIG. 9 COMPARISON OF TSWT AND NPL MODEL PRESSURE DISTRIBUTIONS : $M_\infty \simeq 0.75$; $\alpha \simeq 2^\circ$

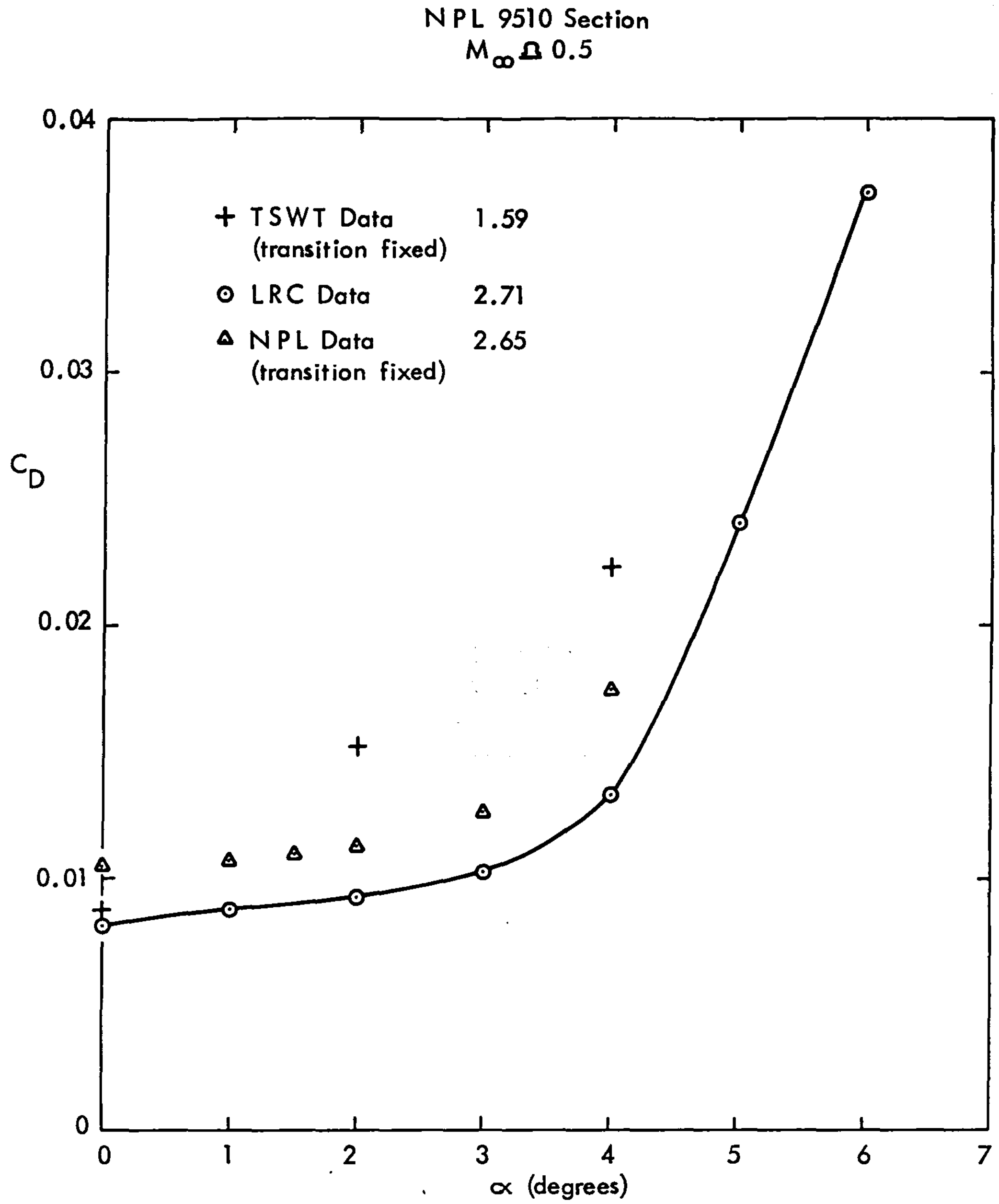


FIG. 10a VARIATION OF DRAG COEFFICIENT WITH ANGLE OF ATTACK: $M_\infty \approx 0.5$

N PL 9510 Section
$M_\infty \cong 0.6$

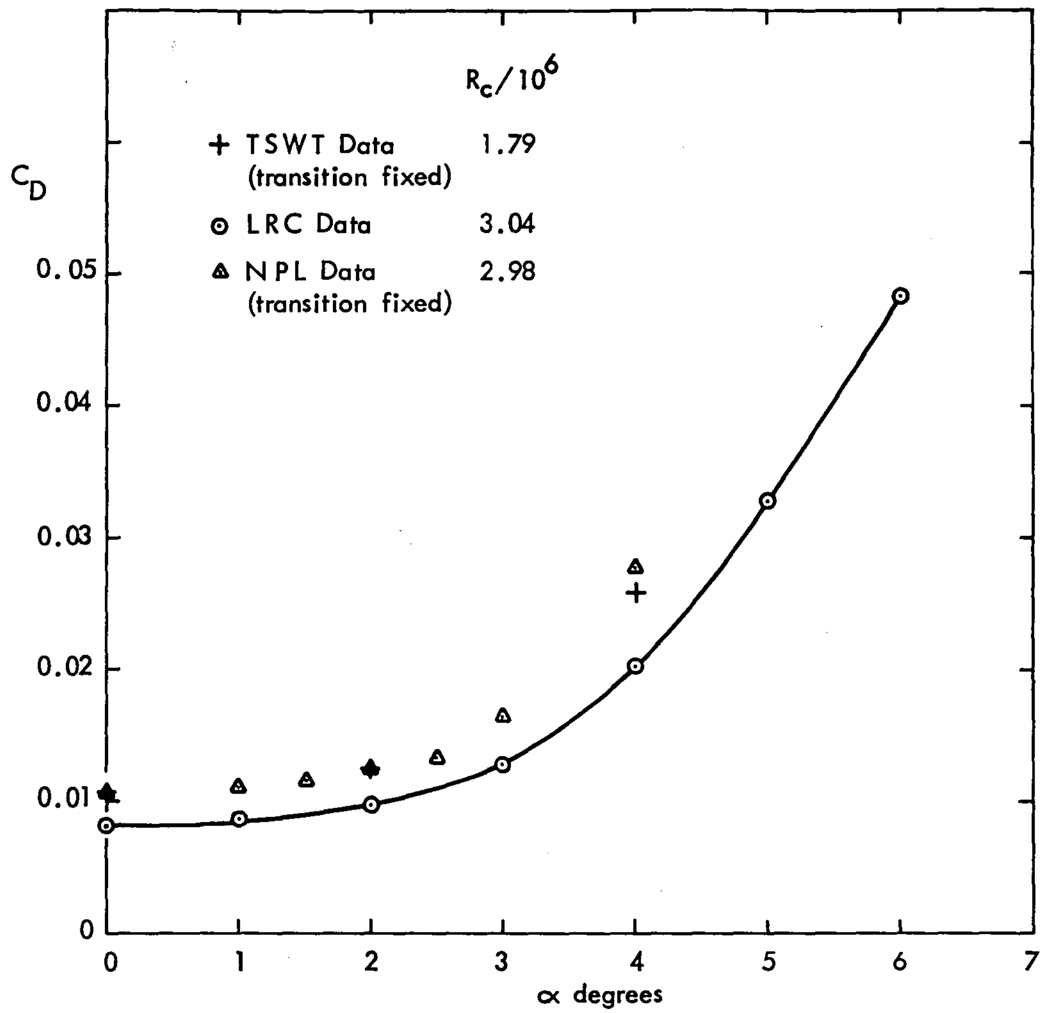


FIG. 10b VARIATION OF DRAG COEFFICIENT WITH ANGLE OF ATTACK: $M_\infty \cong 0.6$

N PL 9510 Section
$M_\infty \simeq 0.7$

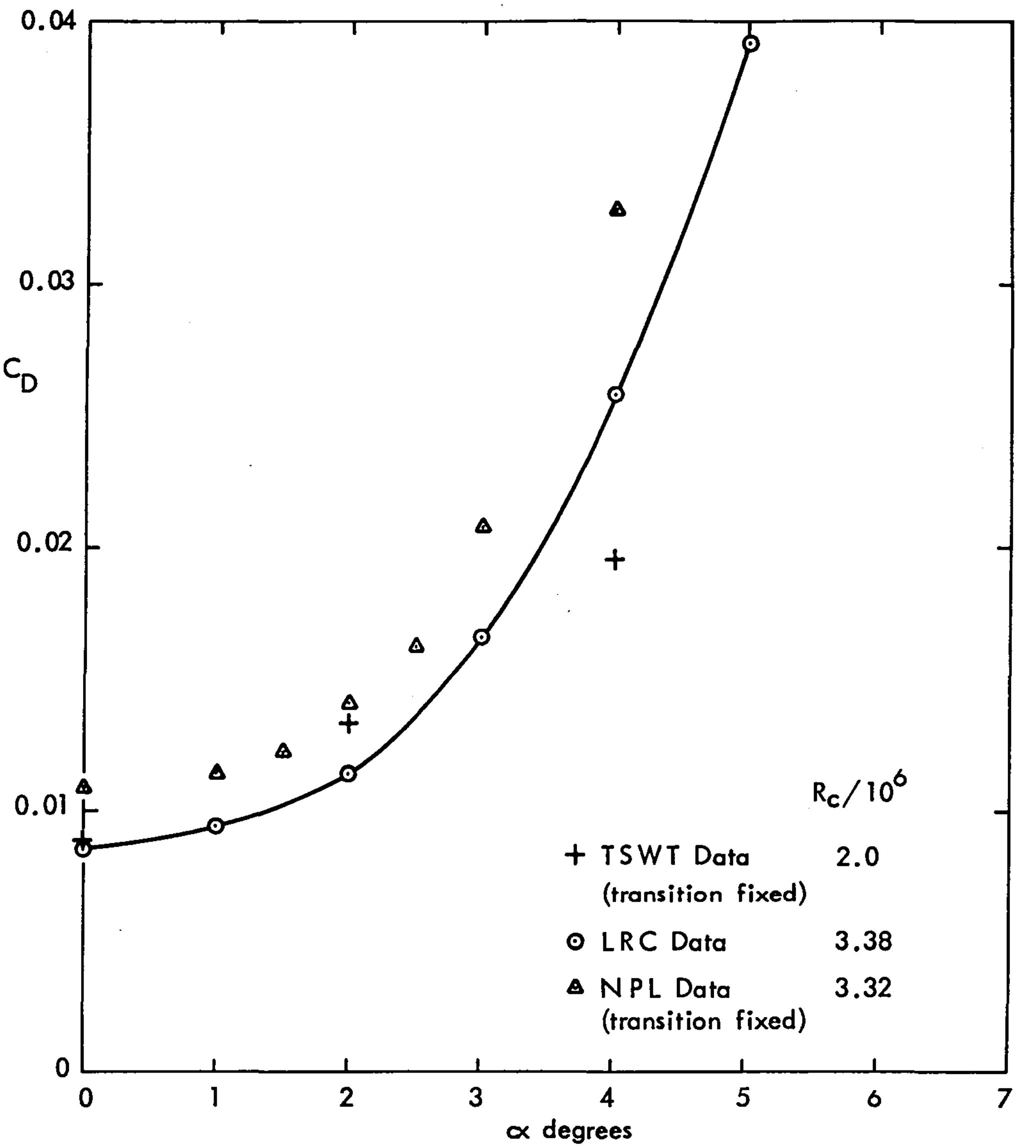


FIG. 10c VARIATION OF DRAG COEFFICIENT WITH ANGLE OF ATTACK: $M_\infty \simeq 0.7$

NPL 9510 Section
$M_{\infty} \simeq 0.75$

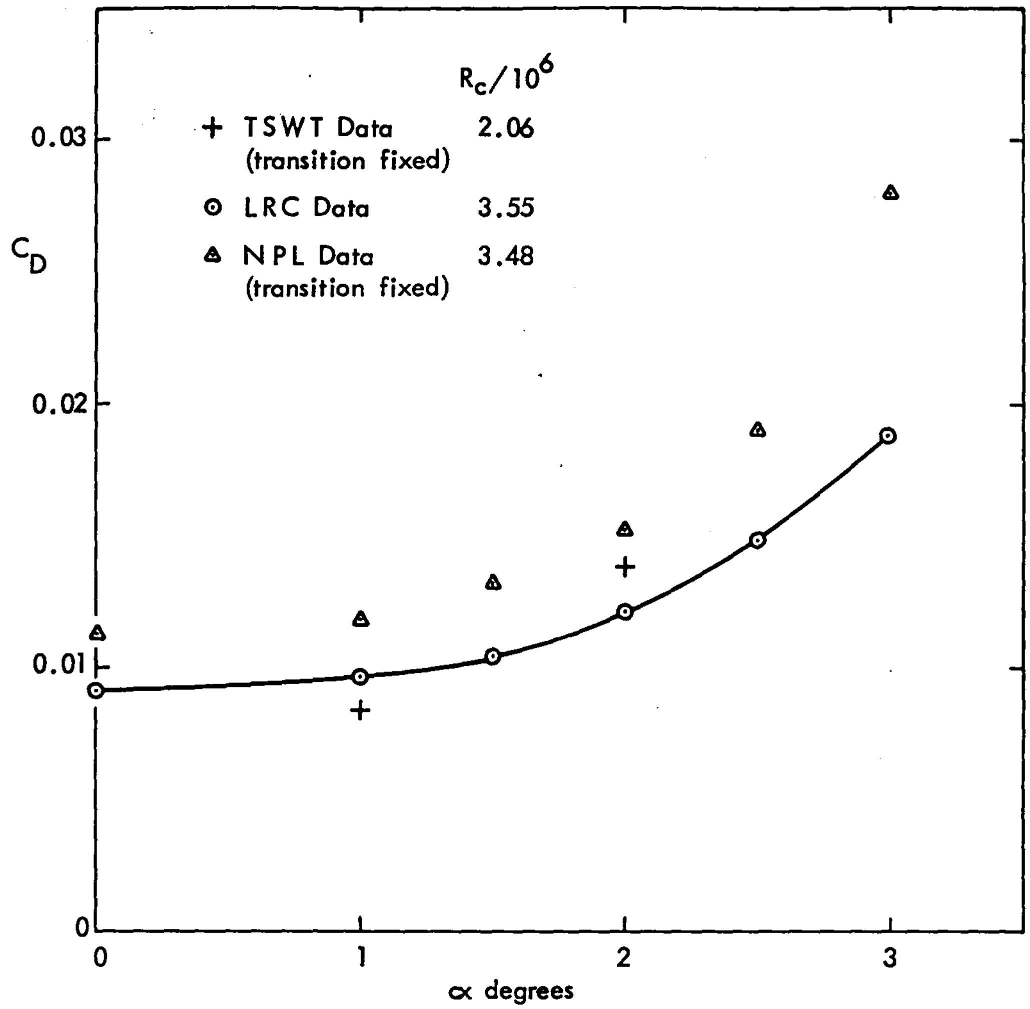


FIG. 10d VARIATION OF DRAG COEFFICIENT WITH ANGLE OF ATTACK: $M_{\infty} \simeq 0.75$

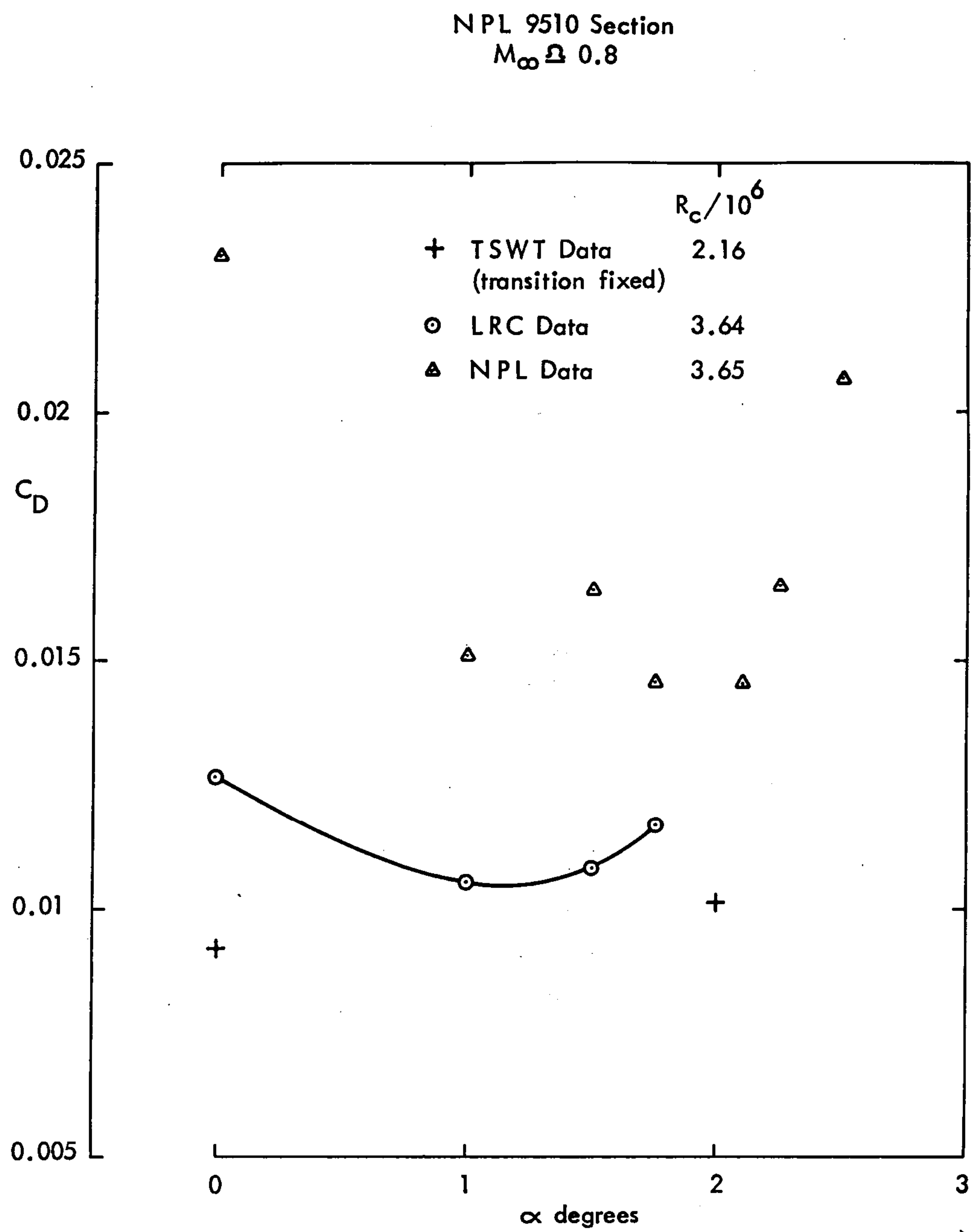


FIG. 10e VARIATION OF DRAG COEFFICIENT WITH ANGLE OF ATTACK: $M_{\infty} \simeq 0.8$

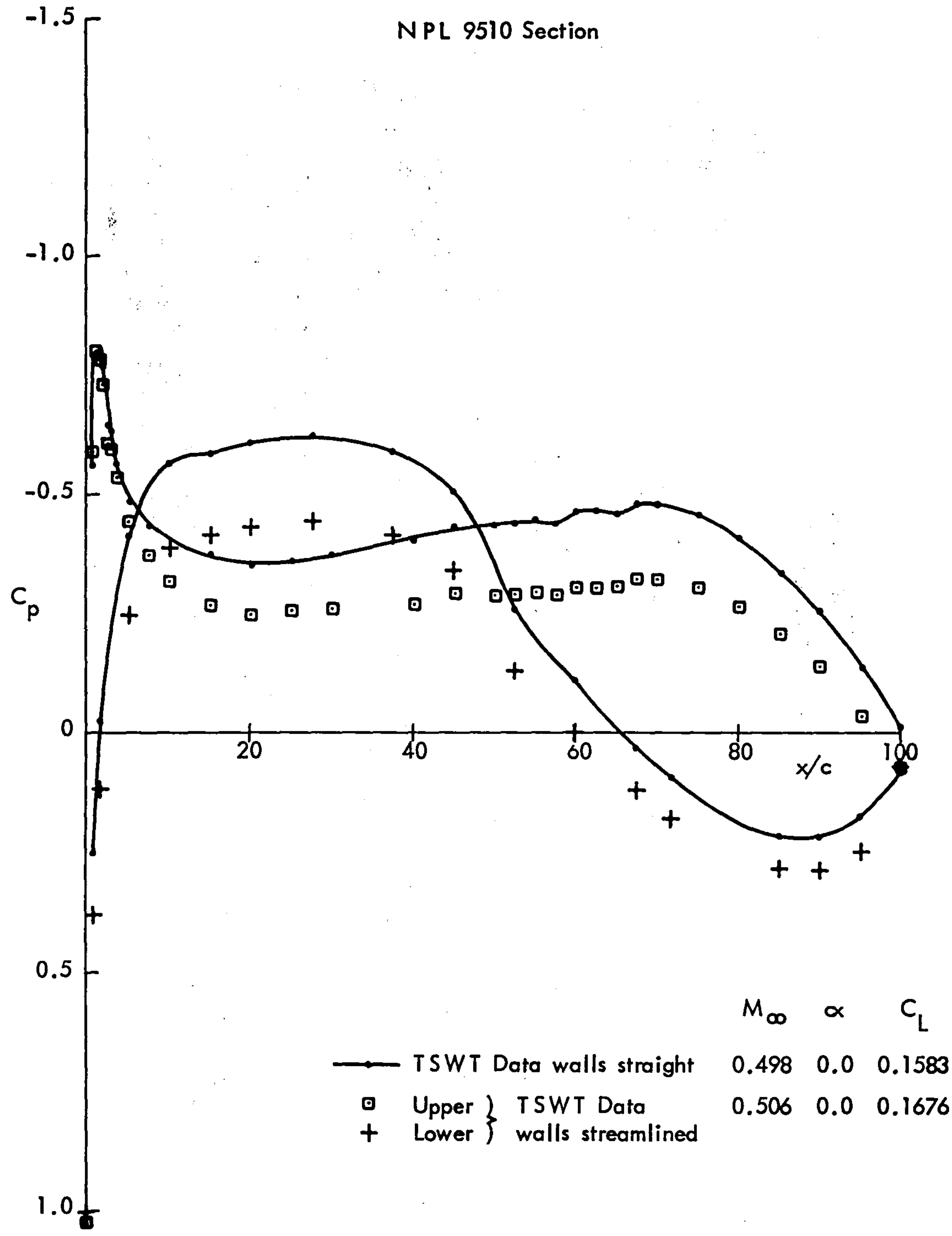


FIG. 11 COMPARISON OF AIRFOIL PRESSURE DISTRIBUTIONS WITH THE WALLS SET STRAIGHT AND STREAMLINED: $M_\infty \simeq 0.5$; $\alpha = 0°$

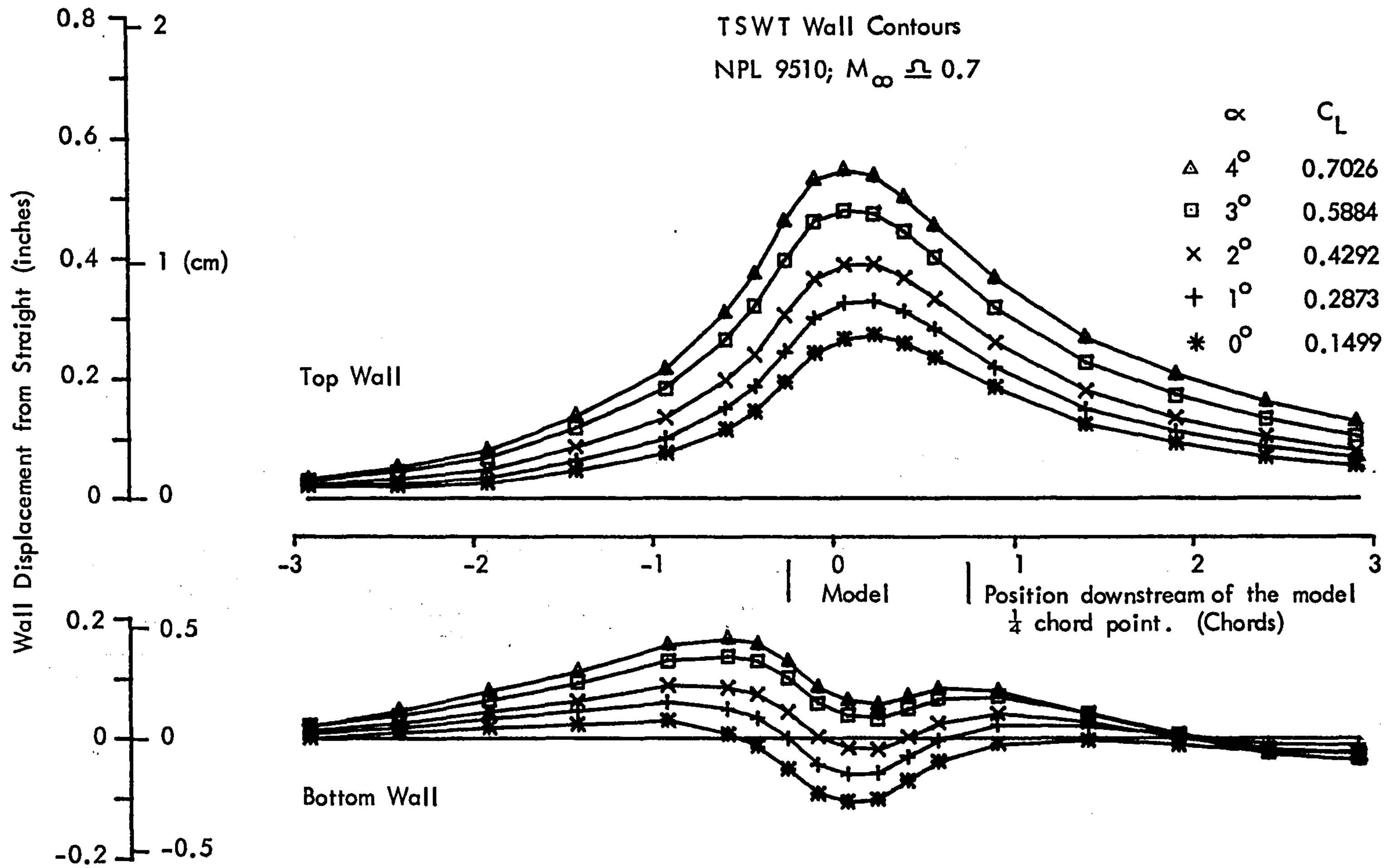


FIG. 12 WALL CONTOURS FOR VARYING ANGLE OF ATTACK AT $M_\infty \simeq 0.7$

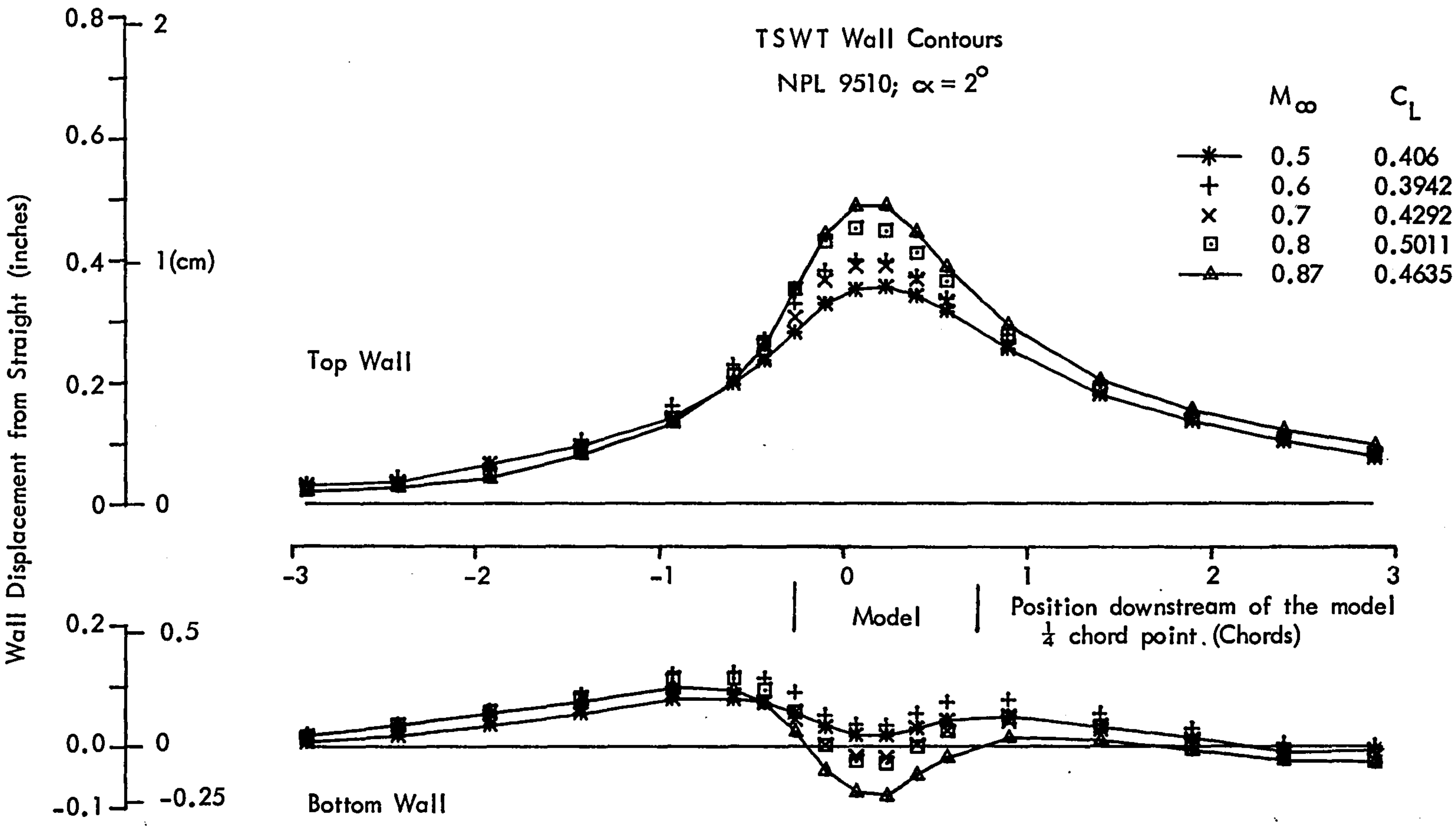


FIG. 13 WALL CONTOUR VARIATIONS WITH MACH NUMBER, α = 2°

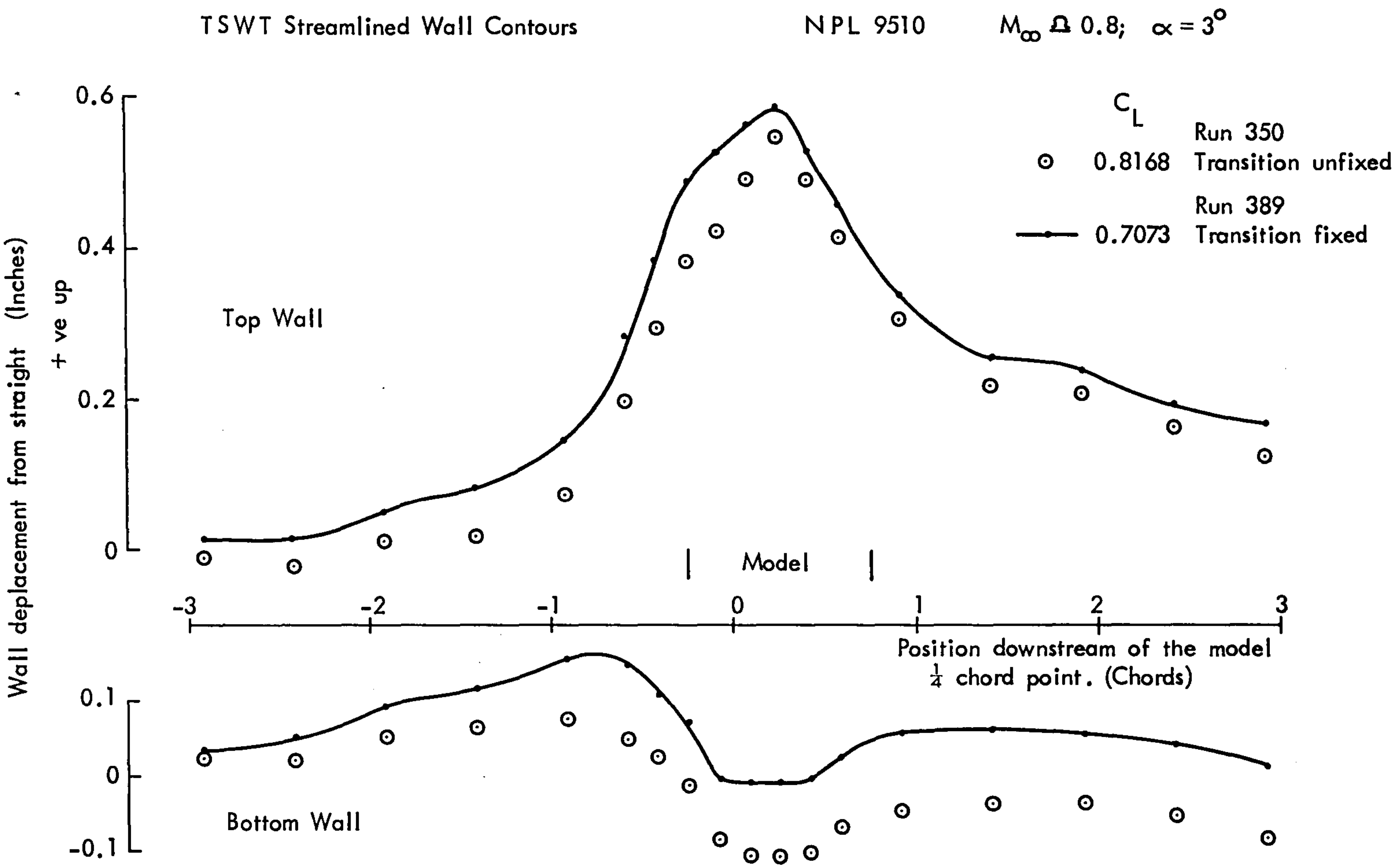


FIG. 14 COMPARISON OF TSWT WALL CONTOURS WITH MODEL TRANSITION FIXED AND UNFIXED

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15. Supplementary Notes Langley Technical Monitor: Charles L. Ladson
This is a progress report 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.

16. Abstract

Further work to validate the flexible wall technique in two-dimensional testing has been carried out with the Transonic Self-Streamlining Wind Tunnel (TSWT) using a cambered NPL 9510 section, larger and perhaps of more challenging design than the NACA 0012-64 section previously tested. Model data on lift and drag were obtained over a Mach number range up to 0.87 and at angles of attack from zero to 6^{o}. Results taken with the walls streamlined were compared with two sources of reference data obtained in conventional slotted walled transonic test sections. The reference data cannot be considered interference free, but is the best currently available at low Reynold's numbers, and has to provide a basis for assessing the quality of TSWT data. There were 52 runs of the test section in carrying out this program. Some of the streamlining cycles were performed using an automated wall control system linked to a mini-computer. These runs provided further useful TSWT operational experience with a larger model than previously tested. Limits to both test Mach number and model angle of attack were found. NPL 9510 data from TSWT is presented as a library of numerical and graphical information which may prove useful to others engaged in the evaluation, design and use of transonic flexible walled test sections.

17. Key Words (Suggested by Author(s)) Aerodynamics, Airfoils, Transonic Wind Tunnels, Adaptive Walls	18. Distribution Statement Unclassified - Unlimited; Star Category - 02

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