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A BRIEF DESCRIPTION OF THE JAMESON-CAUGHEY NYU TRANSONIC SWEPT-WING COMPUTER PROGRAM - FLO 22

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Perry A. Newman, and Ruby M. Davis

December 1976

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16. Abstract

Prof. Antony Jameson (NYU) and Prof. Dave Caughey (Cornell) have developed this computer program for analyzing inviscid, isentropic, transonic flow past 3-D swept-wing configurations. This work was done at the Courant Institute of Mathematical Sciences, New York University, under NASA Grants NGR-33-016-167 and NGR-33-016-201. Some basic aspects of the program are: The free-stream Mach number is restricted only by the isentropic assumption. Weak shock waves are automatically located where ever they occur in the flow. The finite-difference form of the full equation for the velocity potential is solved by the method of relaxation, after the flow exterior to the airfoil is mapped to the upper half plane. The mapping procedure allows exact satisfaction of the boundary conditions and use of supersonic free stream velocities. The finite difference operator is "locally rotated" in supersonic flow regions so as to properly account for the domain of dependence. The relaxation algorithm has been stabilized using criteria from a time-like analogy. The brief description contained in this document should enable one to use the program until a formal user's manual is available.

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PREFACE

This document was prepared by the third and fourth authors to serve as an interim user guide for the Jameson-Caughey NYU Transonic Swept-Wing Computer Program - FLO 22. This information is pertinent to the version provided to NASA LaRC in the spring of 1976. The first and second authors are in the process of preparing more extensive documentation on a later version of program FLO 22. This document is intended to meet several immediate demands for use of this computer code in NASA-sponsored programs.

The first two sections entitled "Calculation of the Flow Past a Swept Wing" and "Some Results of Swept Wing Calculations" are sections 5.7 and 5.8 of reference 1 with very minor editing. The last section entitled "Input Description" is an update of reference 2 (an input description for Jameson's yawed wing program - FLO 17) which includes the swept-wing input parameters.

SUMMARY

Prof. Antony Jameson (NYU) and Prof. Dave Caughey (Cornell) have developed this computer program for analyzing inviscid, isentropic, transonic flow past 3-D swept-wing configurations. This work was done at the Courant Institute of Mathematical Sciences, New York University, under NASA Grants NGR-33-016-167 and NGR-33-016-201. Some basic aspects of the program are: The free-stream Mach number is restricted only by the isentropic assumption. Weak shock waves are automatically located where ever they occur in the flow. The finite-difference form of the full equation for the velocity potential is solved by the method of relaxation, after the flow exterior to the airfoil is mapped to the upper half plane. The mapping procedure allows exact satisfaction of the boundary conditions and use of supersonic free stream velocities. The finite difference operator is "locally rotated" in supersonic flow regions so as to properly account for the domain of dependence. The relaxation algorithm has been stabilized using criteria from a time-like analogy. The brief description contained in this document should enable one to use the program until a formal user's manual is available.

CALCULATION OF THE FLOW PAST A SWEPT WING

It is desired to solve the three-dimensional potential flow equation which can be written in quasilinear form as

$$(a^2-u^2)\phi_{xx} + (a^2-v^2)\phi_{yy} + (a^2-w^2)\phi_{zz}$$

$$- 2uv\phi_{xy} - 2vw\phi_{yz} - 2uw\phi_{xz} = 0 \qquad (1)$$

where u, v and w are the velocity components and a is the local speed of sound. The singularity at infinity in the velocity potential is removed by introducing a reduced potential

$$G = \phi - x \cos \alpha - y \sin \alpha \qquad (2)$$

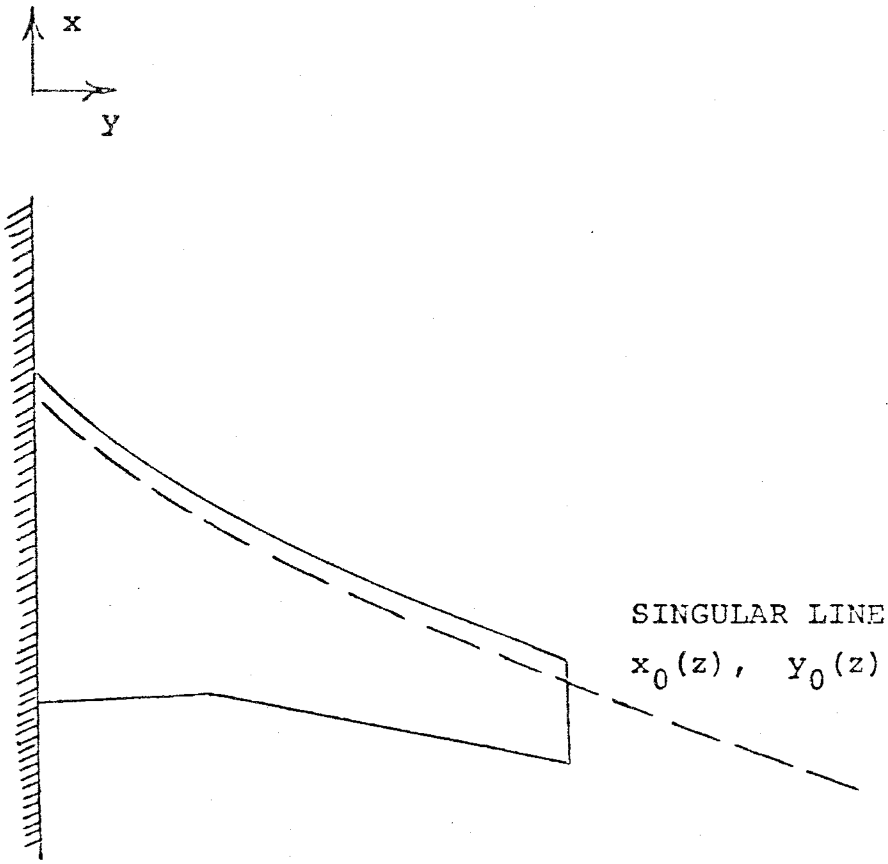
where α is the angle of attack.

In the case of a lifting flow the velocity potential is discontinuous across the vortex sheet trailing behind the wing. Roll up of the vortex sheet will be ignored: the conditions to be satisfied at the surface in which the vortex sheet is assumed to lie are that the jump Γ in the potential is constant along lines parallel to the free stream, and that the normal component of velocity is continuous through the sheet. At infinity the flow is undisturbed except in the Trefftz plane far downstream where there will be a two dimensional flow induced by the vortex sheet.

The construction of a satisfactory curvilinear coordinate system to suit the geometry of the configuration is one of the most difficult aspects of the three dimensional problem. Here nonorthogonal coordinates will be generated by a sequence of elementary transformations. First parabolic coordinates are introduced in planes containing the wing section by the square root transformation

$$X_1 + iY_1 = [x - x_0(z) + i(y-y_0(z))]^{1/2} , \quad Z_1 = z , \tag{3}$$

where z is the spanwise coordinate, and x_0 and y_0 define a singular line of the coordinate system located just inside the leading edge (see Figures 1 and 2). The effect of this transformation is to unwrap the wing to form a shallow bump



PLANE OF SYMMETRY

FIGURE 1. CONFIGURATION OF SWEPT WING

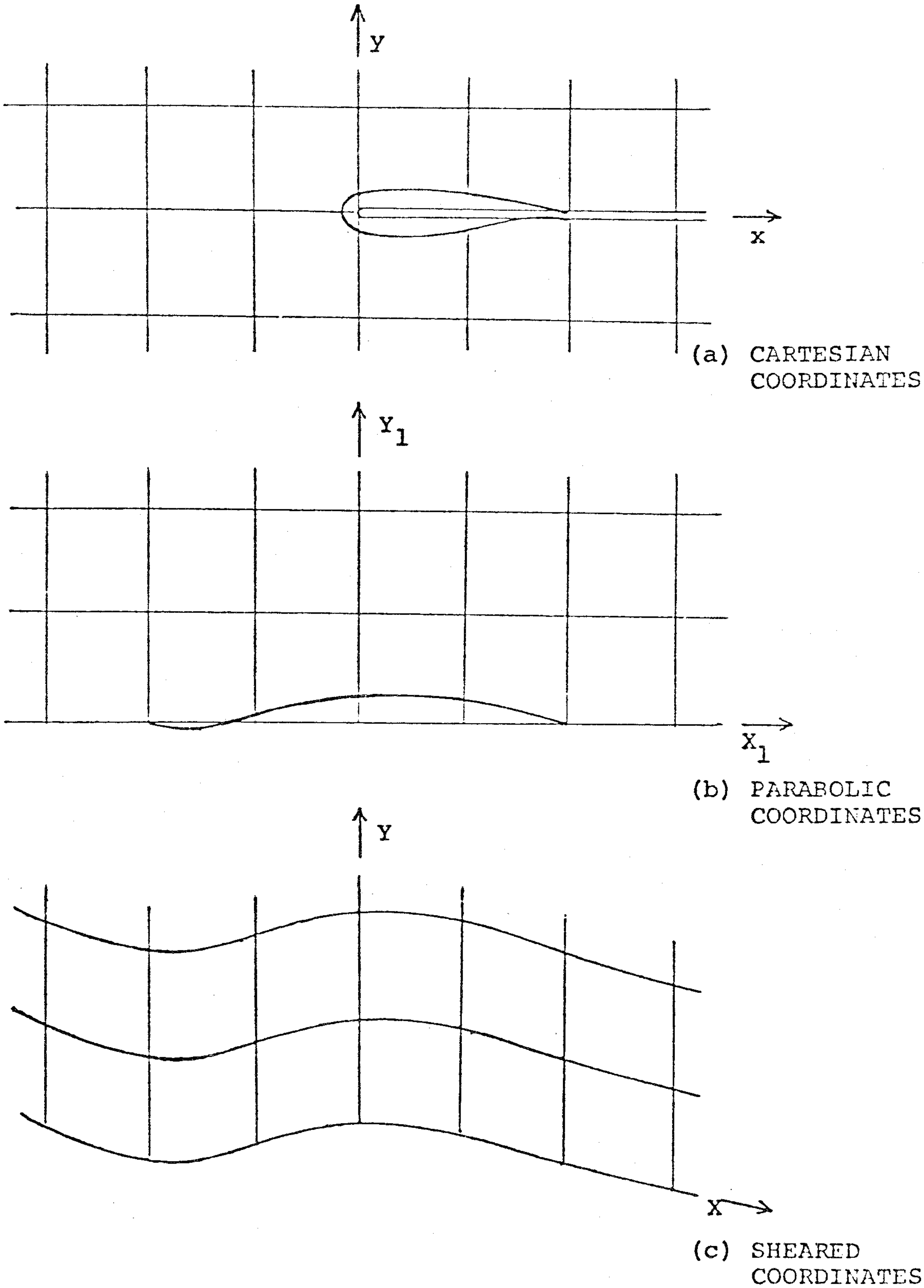


FIGURE 2. CONSTRUCTION OF COORDINATE SYSTEM FOR SWEPT WING CALCULATION

$$Y_1 = S(X_1, Z_1) \tag{4}$$

Then a shearing transformation is used

$$X = X_1 , \quad Y = Y_1 - S(X_1, Z_1) , \quad Z = Z_1 \tag{5}$$

to map the wing surface to a coordinate surface. Finally, in order to obtain a finite computational domain X, Y and Z are replaced by stretched coordinates $\bar{X}$, $\bar{Y}$ and $\bar{Z}$. The stretching used in the present computer program is to set $X = \bar{X}$ in an inner domain $-\bar{X}_m \leq \bar{X} \leq \bar{X}_m$, and to set

$$X = \bar{X}_m + (\bar{X} - \bar{X}_m) / \left\{1 - \left(\frac{\bar{X}-\bar{X}_m}{1-\bar{X}_m}\right)^2\right\}^{\alpha} \tag{6}$$

when $\bar{X} > \bar{X}_m$, with a corresponding formula when $\bar{X} < -\bar{X}_m$, so that $X = \pm\infty$ when $\bar{X} = \pm 1$. Typically the parameter α has the value 1/2. Similar stretchings are used for Y and Z.

The vortex sheet is assumed to coincide with the cut behind the singular line which is opened up by the square root transformation (Eq. (3)). Thus a jump Γ is introduced in the potential between corresponding points representing the two sides of the vortex sheet. A complication is caused by the continuation of the cut beyond the wing. Points on the two sides of the cut must be identified as the same point in the physical space. Also a special form of the equations must be used at points lying on the singular line beyond the wing. At these points the equation to be satisfied reduces to the two dimensional Laplace equation in the X_1 and Y_1 coordinates. An advantage of the square root transformation (Eq. (3)) is that it collapses the height of the disturbance due to the vortex sheet to zero in the parabolic coordinate system at points far downstream, where X_1 approaches infinity, with the result that the far field boundary condition is simply

$$G = 0 \tag{7}$$

The final equation for the reduced potential G contains numerous terms. In order to construct a rotated difference scheme with a proper upwind bias at supersonic points it is only necessary, however, to consider the principal part, consisting of the terms containing the second derivatives of G. Suppose that equation (1) is written in the canonical form

$$(a^2 - q^2)\phi_{ss} + a^2(\Delta\phi - \phi_{ss}) = 0 \tag{8}$$

where Δ is the Laplacian operator $\frac{\partial^2}{\partial x^2} + \frac{\partial^2}{\partial y^2} + \frac{\partial^2}{\partial z^2}$ and ϕ_{ss} is the streamwise second derivative in a Cartesian coordinate system locally aligned with the flow

$$\phi_{ss} = \frac{1}{q^2}\left(u^2\phi_{xx} + v^2\phi_{yy} + w^2\phi_{zz} + 2uv\phi_{xy} + 2vw\phi_{yz} + 2uw\phi_{xz}\right). \tag{9}$$

Then at supersonic points upwind difference formulas are used for all second derivatives of G arising from the transformation of ϕ_{ss} into the curvilinear coordinate system, and central difference formulas for all second derivatives of G arising from the transformation of $\Delta\phi - \phi_{ss}$. The formulation in terms of the Laplacian avoids the need to determine explicitly a pair of local coordinate directions normal to the stream direction.

The difference equations are solved by relaxation, with care taken to make sure that at supersonic points the equivalent time dependent equation is compatible with the steady state equation. If m and n are coordinates in a plane normal to the streamwise direction s, and M is the local Mach number q/a, the equivalent time dependent equation can be written in the form

$$(M^2-1)\phi_{ss} - \phi_{mm} - \phi_{nn} + 2\alpha_1\phi_{st} + 2\alpha_2\phi_{mt} + 2\alpha_3\phi_{nt} + \gamma\phi_t = 0 \qquad (10)$$

where the coefficients α_1, α_2 and α_3 depend on the split between new and old values used in the relaxation scheme. To make sure that this is a wave equation with s as the time-like direction, an analysis indicates that the difference formulas should be organized so that

$$\alpha_1 > \sqrt{(M^2-1)(\alpha_2^2+\alpha_3^2)} \qquad (11)$$

Also the balanced coefficient rule should still be applied in the supersonic zone, corresponding to a zero coefficient of ϕ_t in equation (10). It is convenient to solve the equations for the correction to the potential simultaneously along lines, corresponding to a point relaxation process in two dimensions. Any of the coordinate lines can be used for this purpose, the choice being guided by the need to avoid advancing through the supersonic zone in a direction opposed to the flow. In practice it has been found convenient to divide each $\bar{X}$, $\bar{Y}$ plane into three strips, and to march towards the wing surface in each central strip, updating horizontal lines, and then outwards in the left-hand and right-hand strips, updating vertical lines. (See Figure 3.)

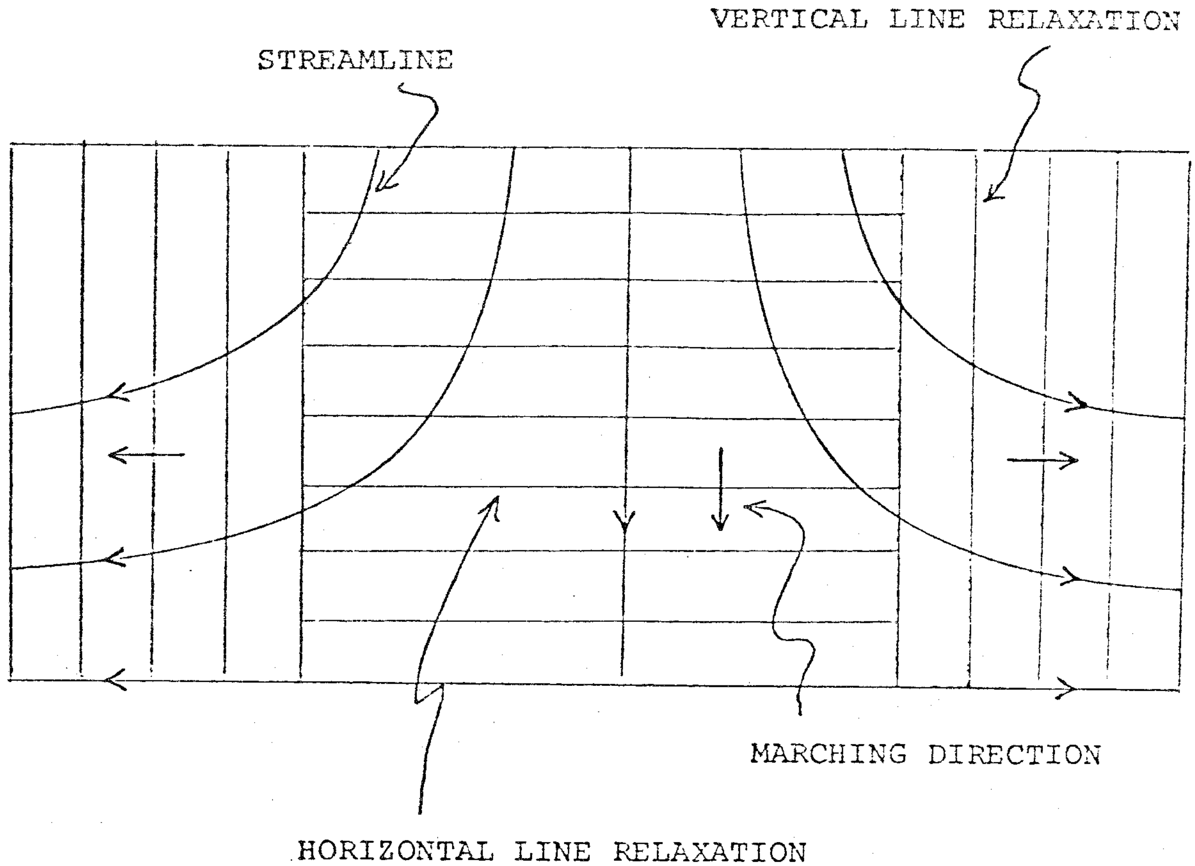


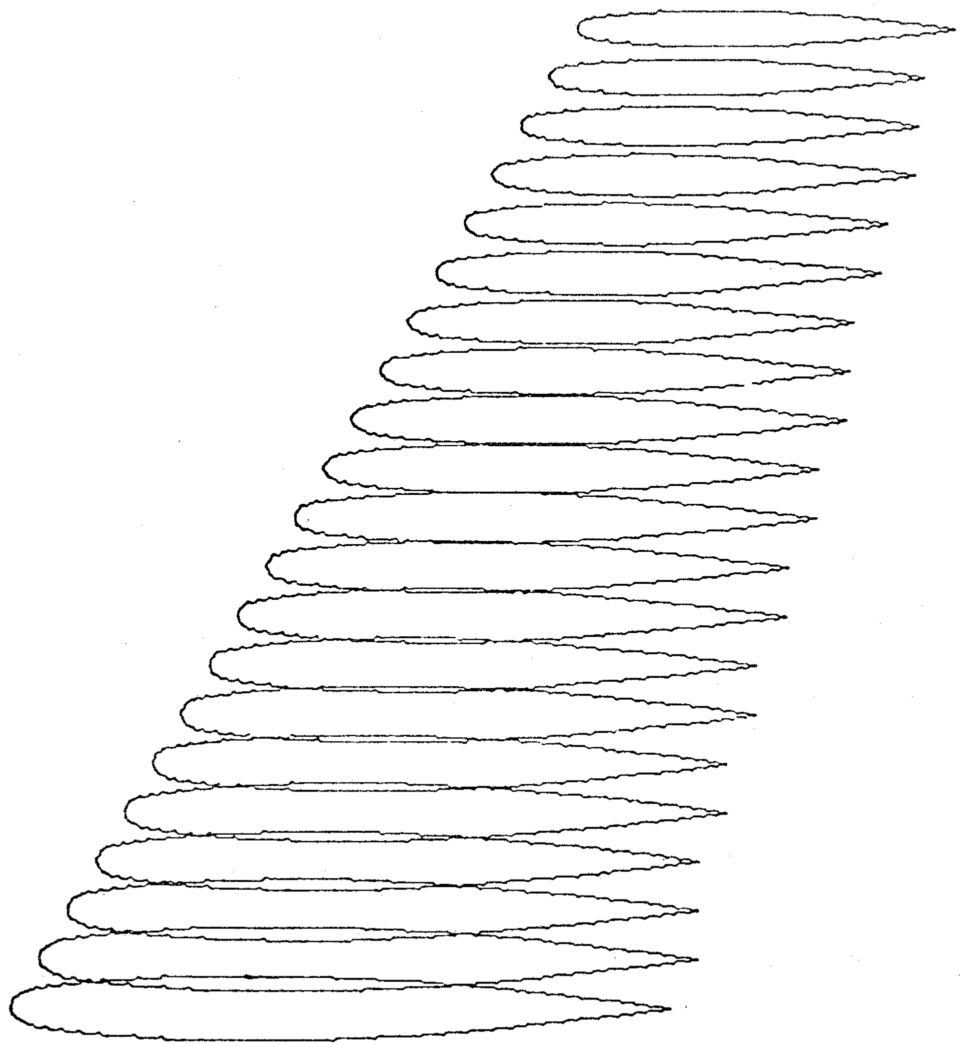
FIGURE 3. MARCHING DIRECTIONS OF RELAXATION SCHEME FOR SWEPT WING CALCULATION

SOME RESULTS OF SWEPT WING CALCULATIONS

Some results of swept wing calculations are presented in Figures 4 and 5. These were calculated on a grid with 144 cells in the chordwise X direction, 24 cells in the normal Y direction and 32 cells in the spanwise Z direction, calling for the solution of the difference equations at 109824 mesh points. In each case the interpolated result of a preliminary calculation on a 72×12×16 grid was used to provide the starting guess: 200 cycles were used on the coarse grid followed by 100 cycles on the fine grid. Such a calculation requires about 80 minutes on a CDC 6600 (which should be reduced to about 20 minutes on a CDC 7600).

Figure 4 shows the result of a calculation for a rather simple wing tested by ONERA, for which experimental data has been published. It can be seen from Figures 4(c) and 4(d) that the agreement with the experimental data is quite good, despite the fact that no attempt was made to allow for viscous effects. As in the case of the two dimensional calculations, the nonconservative difference scheme introduces a source distribution over the shock surfaces, causing a displacement of the streamlines and a forward shift in the location of the shock waves. Apparently this partially compensates for the absence of a correction for the displacement effect of the boundary layer.

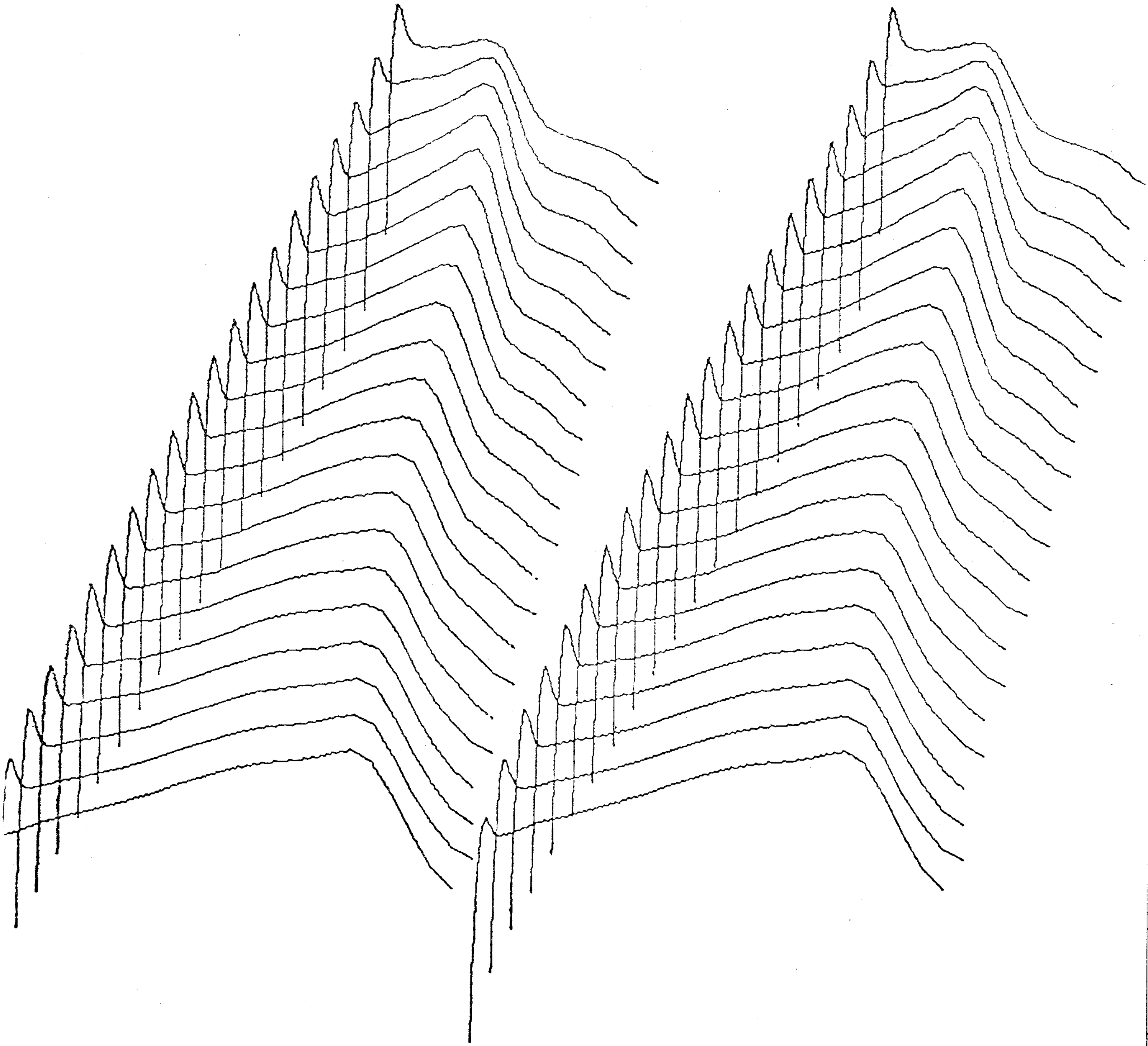
To illustrate the geometric complexity of the configurations which can be treated by the program, Figure 5 shows the results of a calculation for a wing designed and tested by the Douglas Aircraft Company. The wing is a typical design for a long range transport aircraft, with a sweepback of 35 degrees at the leading edge, and a substantial change in the section between the root and tip. The test was of a wing-body combination. In the calculation the wing was extended to the plane of symmetry at the fuselage center line. The calculation shows two shock waves over the inboard part of the wing. The forward shock wave originates from the leading edge at the wing root, and the aft shock wave is roughly normal to the free stream. The two shock waves merge at about the 1/4 span point, forming a triangular shock pattern over the upper surface of the wing. The coalescence of the shock waves can be traced in Figures 5(c) - 5(g), in which the pressure distributions at a sequence of span stations over the inboard part of the wing are plotted separately. The convergence history of this calculation, measured by the largest residual, is shown in Figure 6.



VIEW ØF WING

ØNERA WING M6   L.E. SWEEP 30 DEG   ASPECT RATIØ 3.8
MACH   .923   YAW   0.000   ALPHA   0.000
L/D   -.00   CL   -.0000   CD   .0246

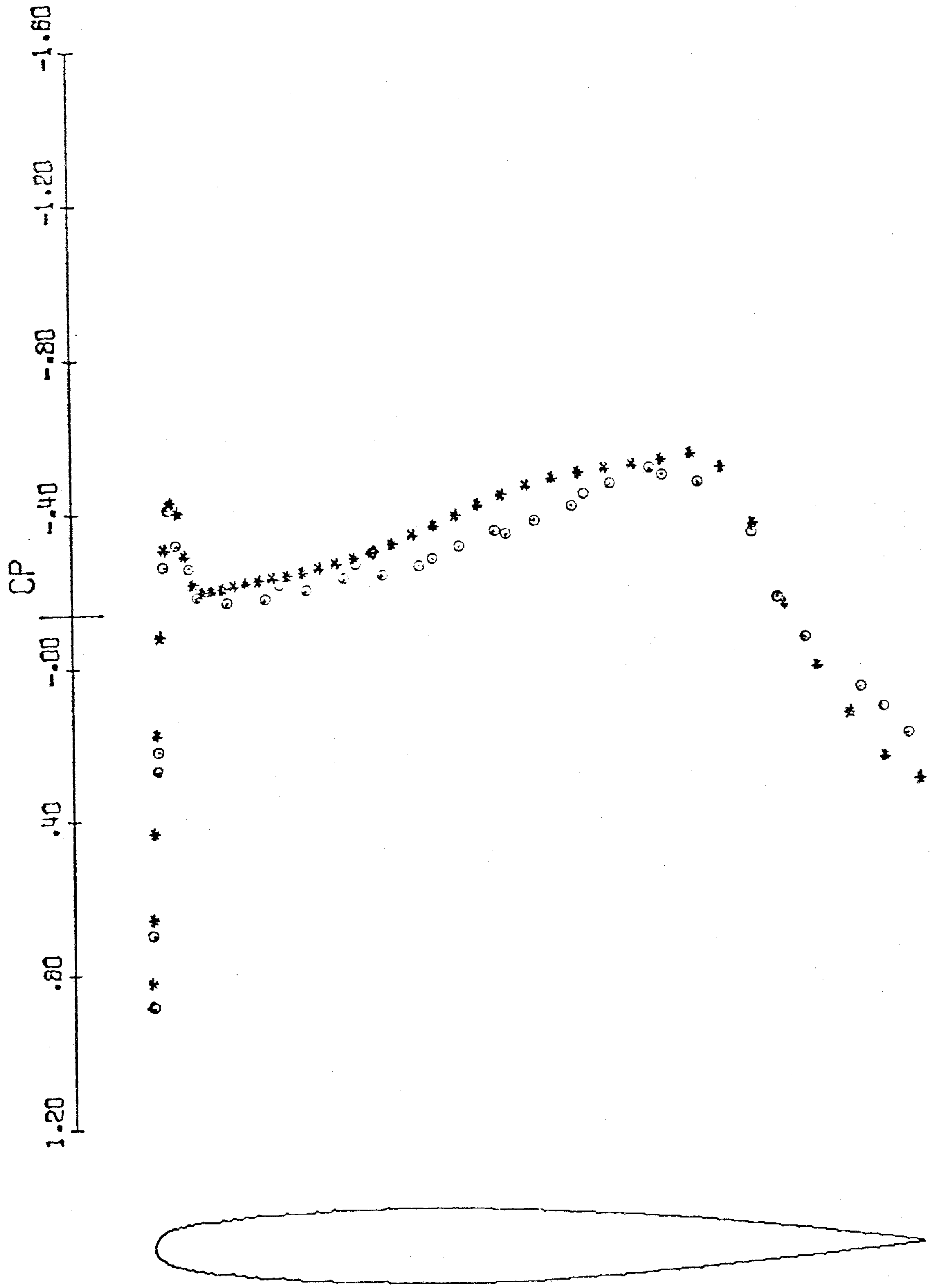
FIGURE 4(A)



’PER SURFACE PRESSURE LØWER SURFACE PRESSURE

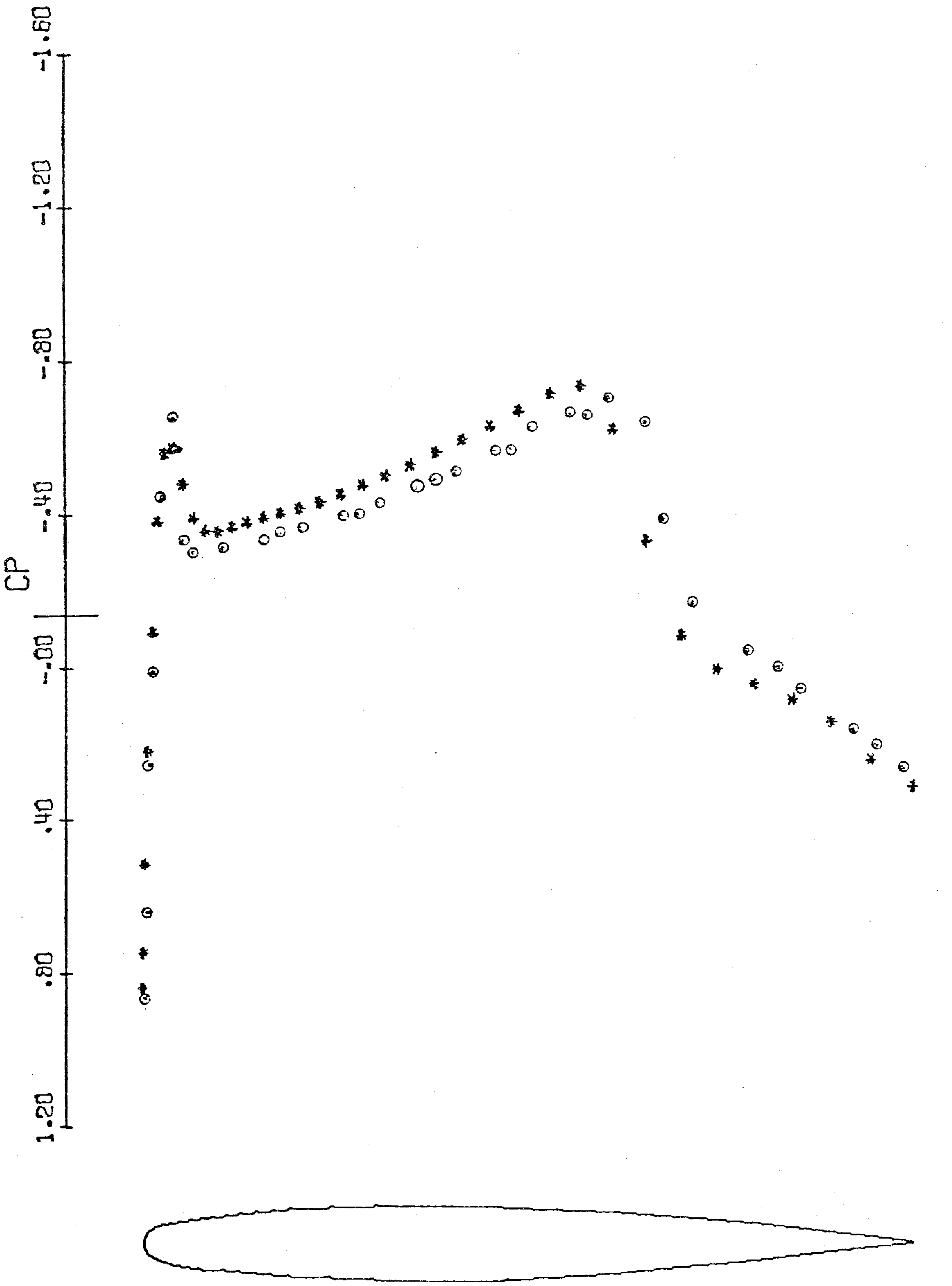
ØNERA WING M6 L.E. SWEEP 30 DEG ASPECT RATIØ 3.8
MACH .923 YAW 0.000 ALPHA 0.000
L/D -.00 CL -.0000 CD .0246

FIGURE 4(B)



ONERA WING M6 L.E. SWEEP 30 DEG ASPECT RATIO 3.8
MACH .923 YAW 0.000 ALPHA 0.000
Z .20 CL - .0000 CD .0242
* THEORY ⊙ EXPERIMENT

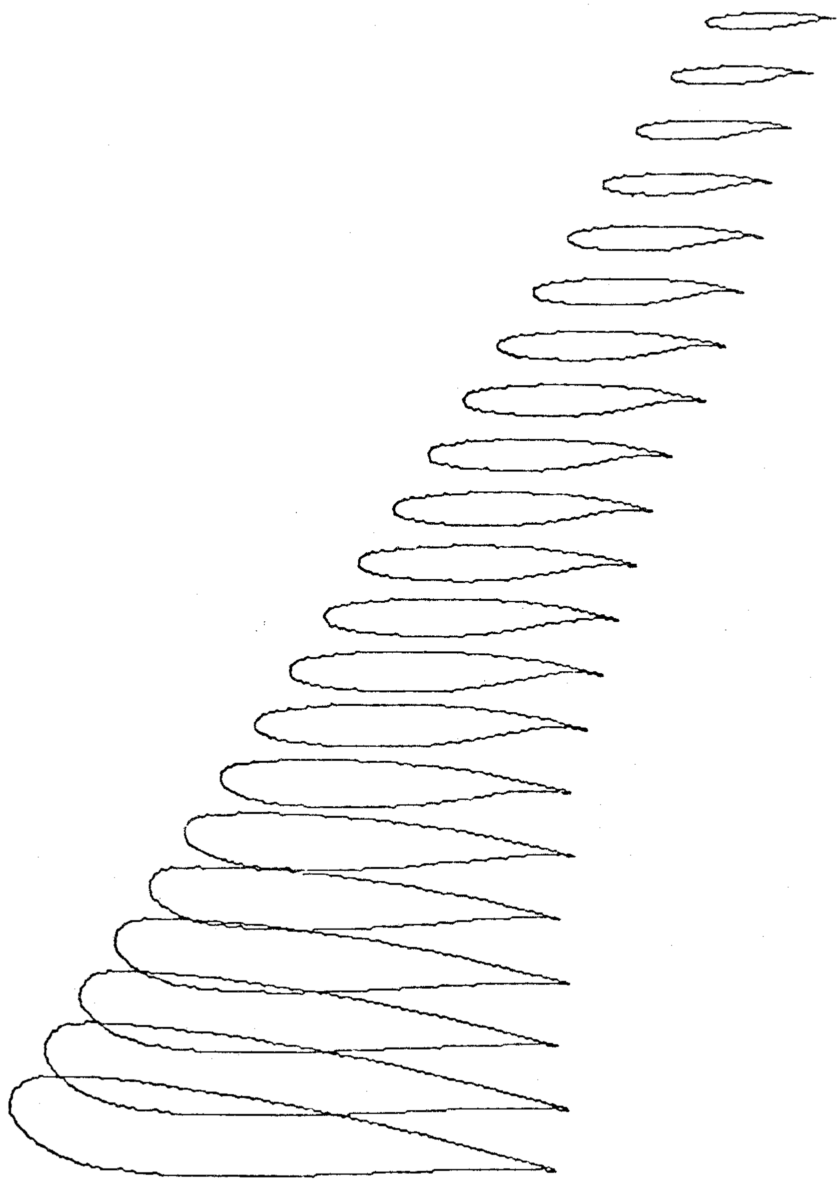
FIGURE 4(C)



ONERA WING M6		L.E. SWEEP 30 DEG		ASPECT RATIO 3.8	
MACH	.923	YAW	0.000	ALPHA	0.000
Z	.80	CL	- .0000	CD	.0009

* THEORY   EXPERIMENT

FIGURE 4(D)

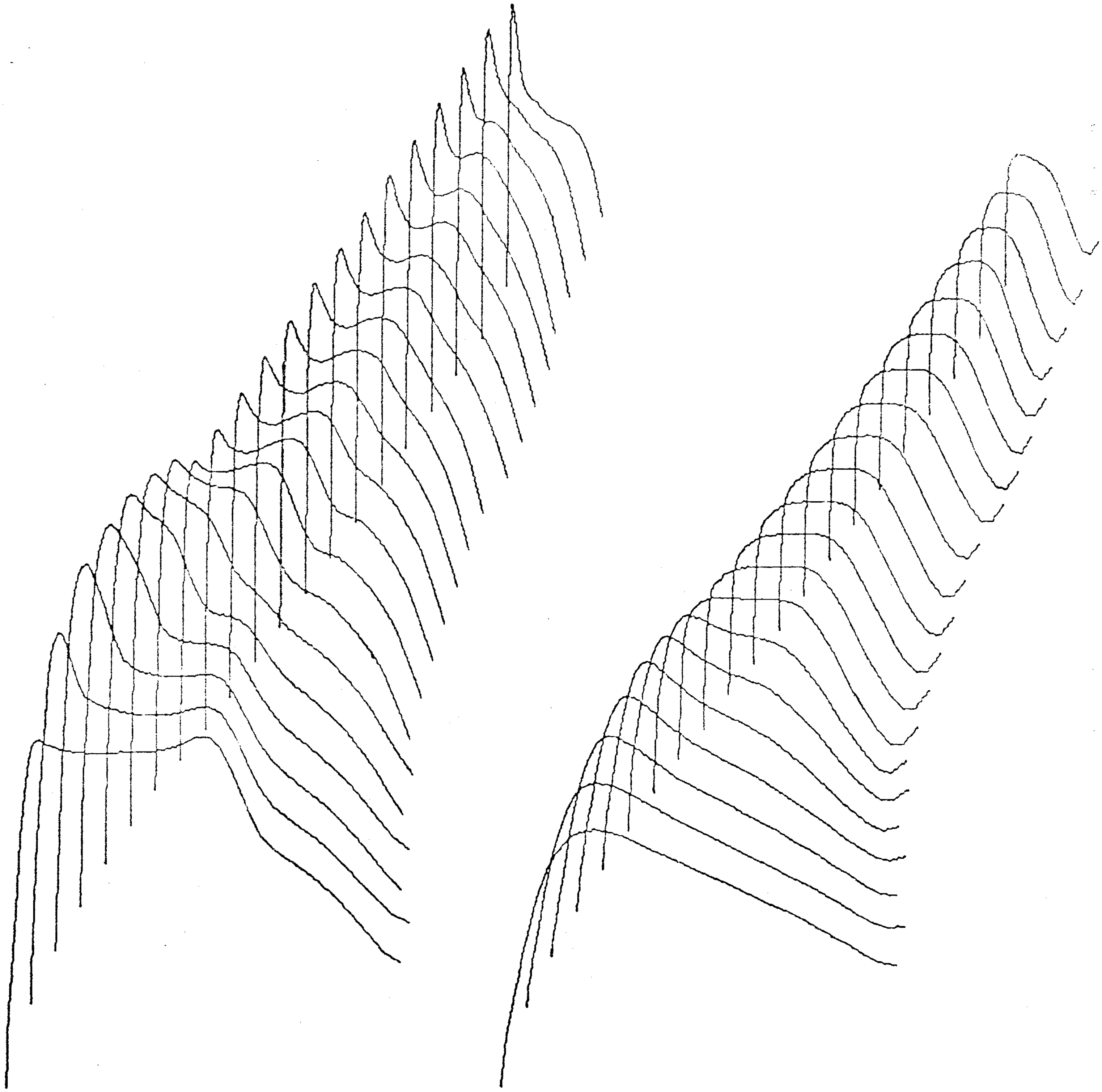


VIEW OF WING

DØUGLAS WING W2 (EXTENDED TO CENTER LINE)

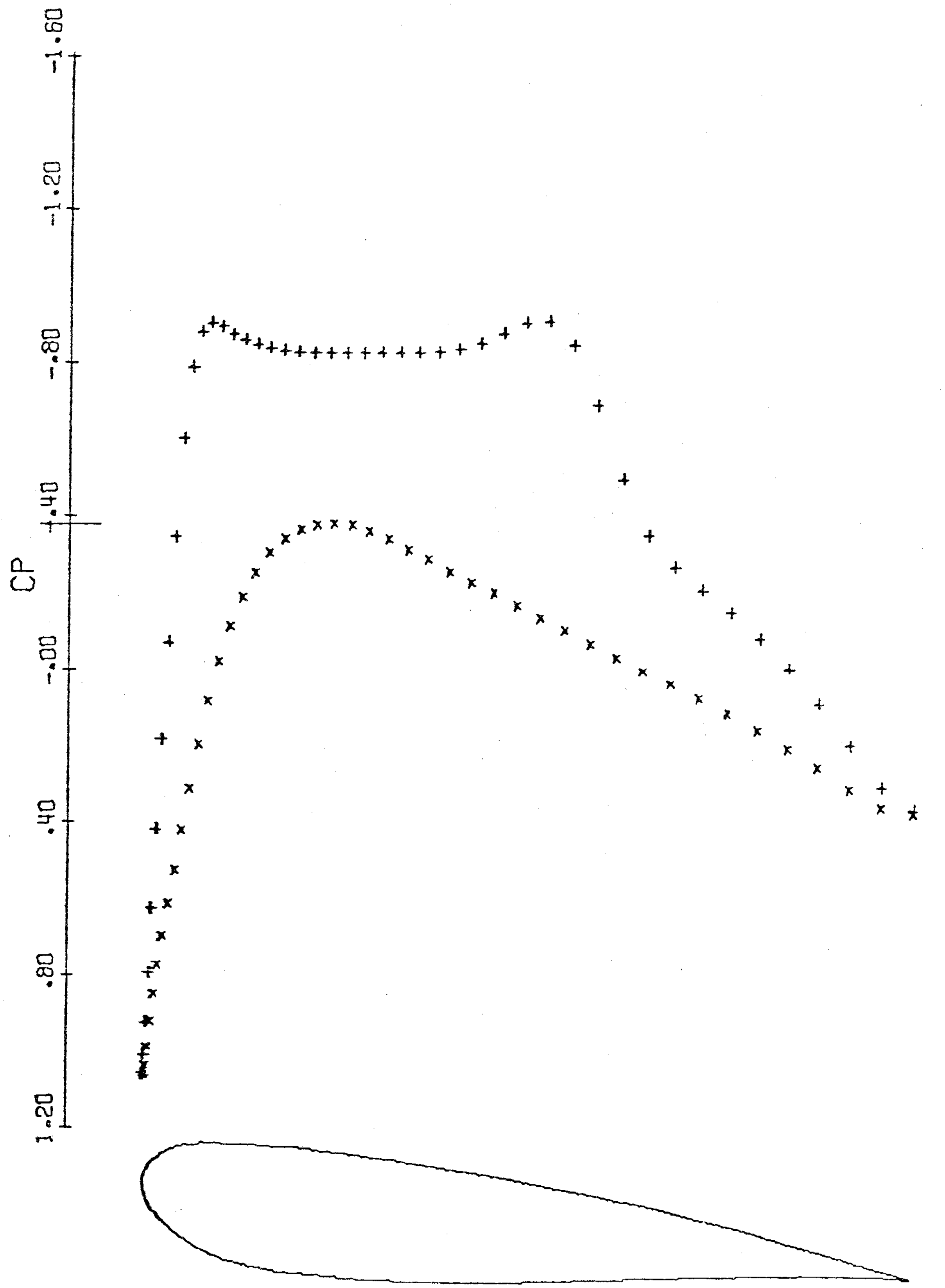
MACH	.819	YAW	0.000	ALPHA	0.000
L/D	23.86	CL	.5882	CD	.0247

FIGURE 5(A)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)
MACH .819 YAW 0.000 ALPHA 0.000
L/D 23.86 CL .5882 CD .0247

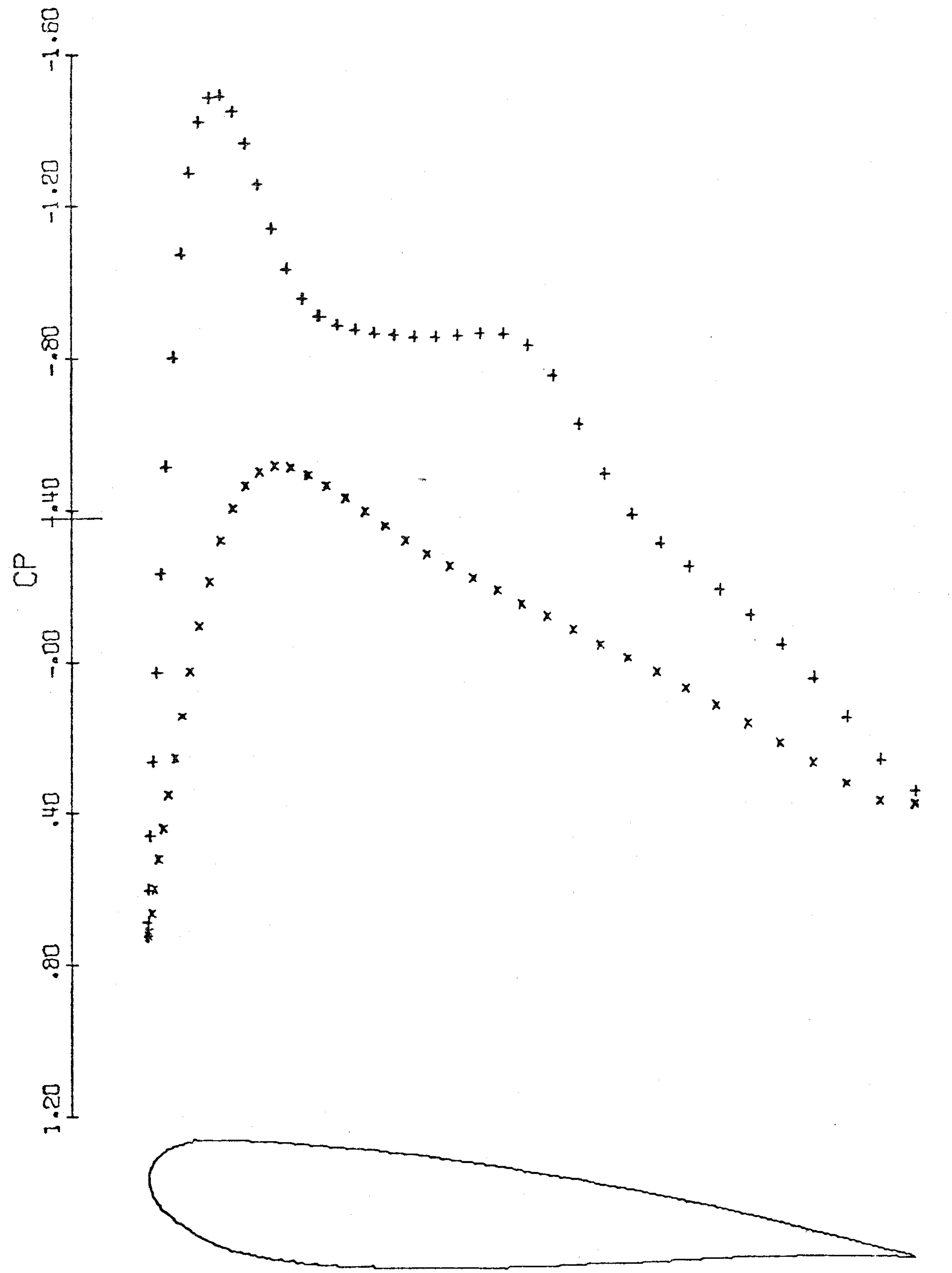
FIGURE 5(B)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)

MACH	.619	YAW	0.000	ALPHA	0.000
Z	0.00	CL	.4688	CD	.1367

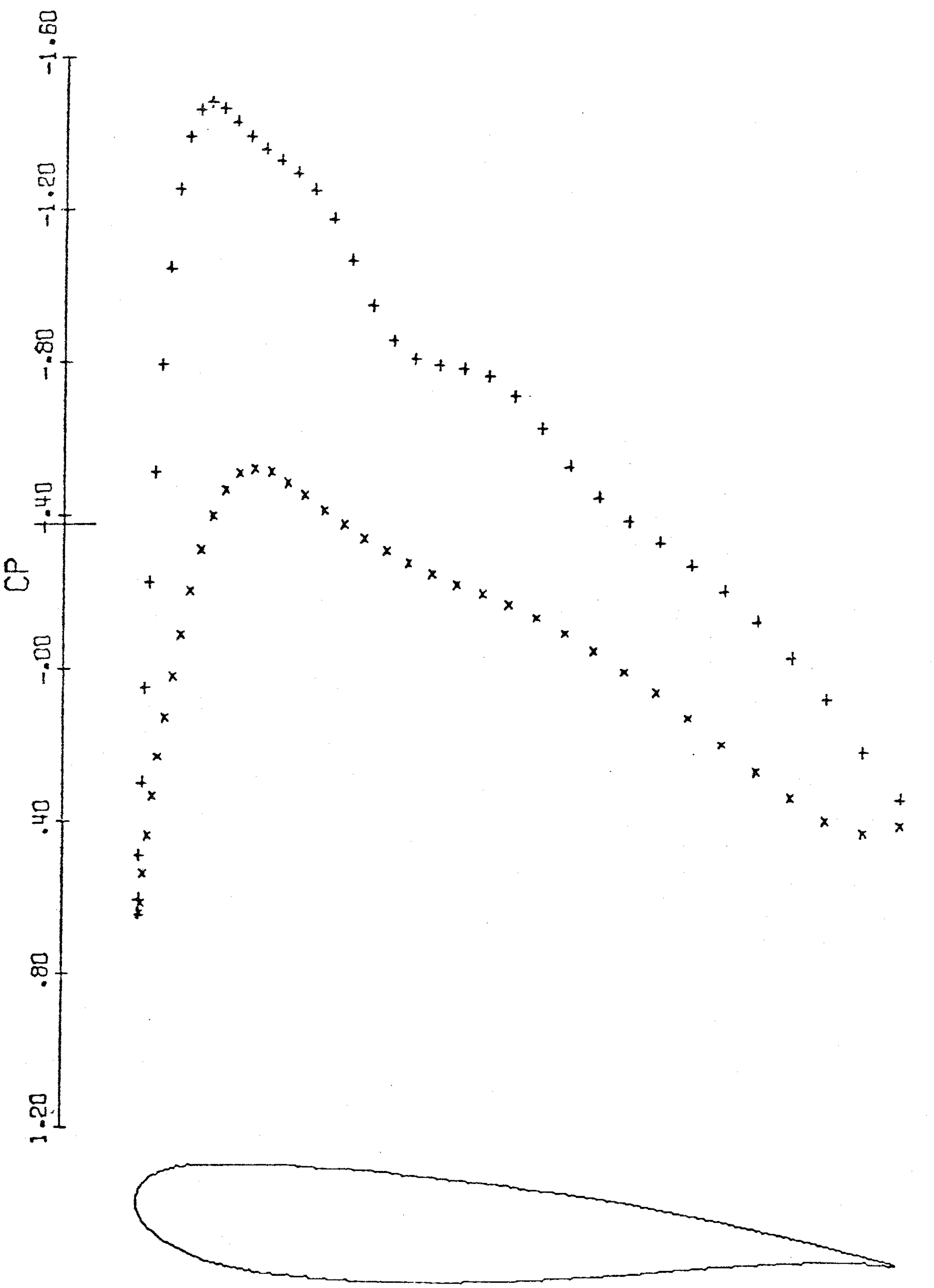
FIGURE 5(C)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)

MACH	.819	YAW	0.000	ALPHA	0.000
Z	2.40	CL	.5060	CD	.0667

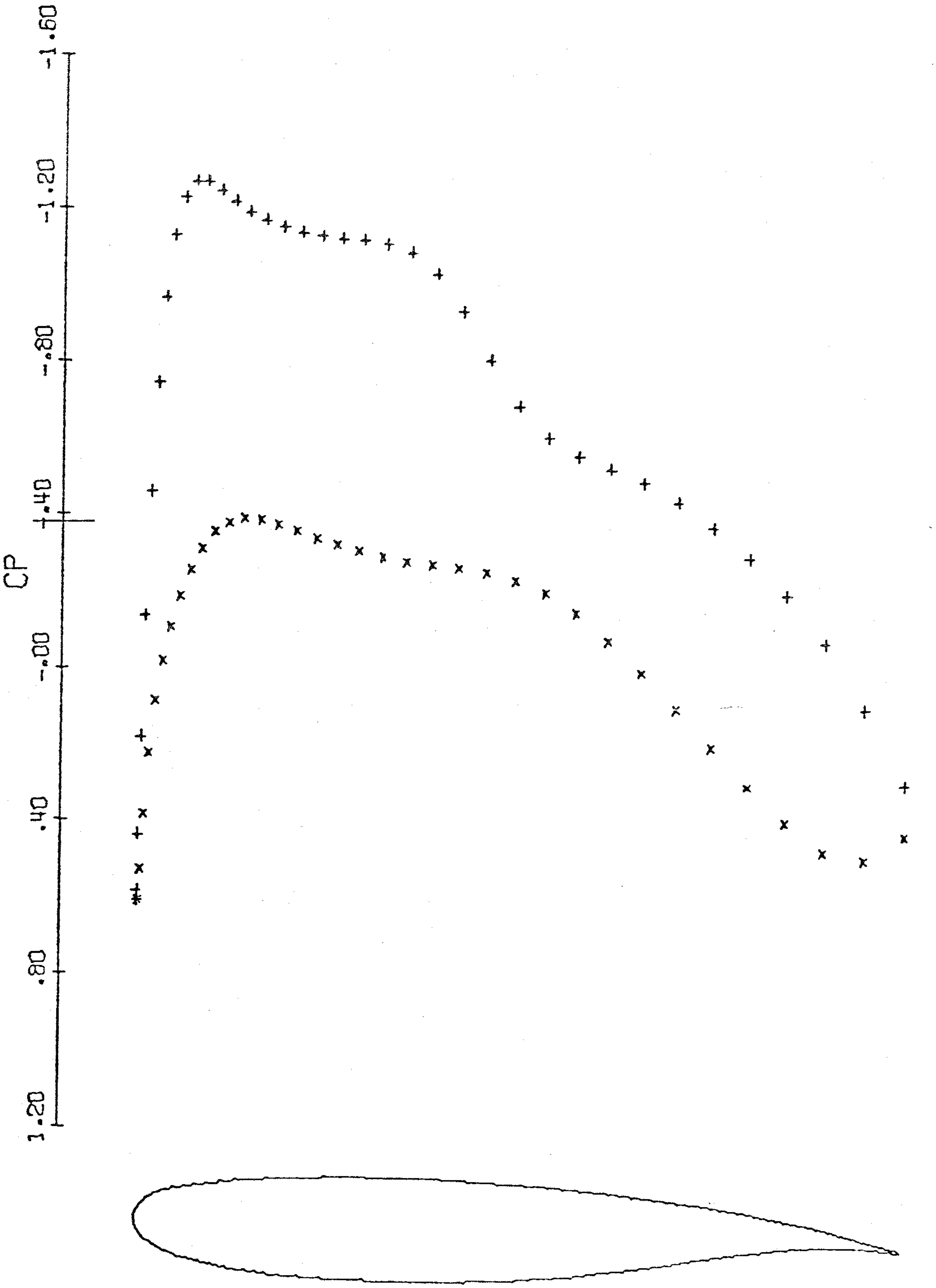
FIGURE 5(D)



DØUGLAS WING W2 (EXTENDED TØ CENTER LINE)

MACH	.819	YAW	0.000	ALPHA	0.000
Z	4.80	CL	.5625	CD	.0301

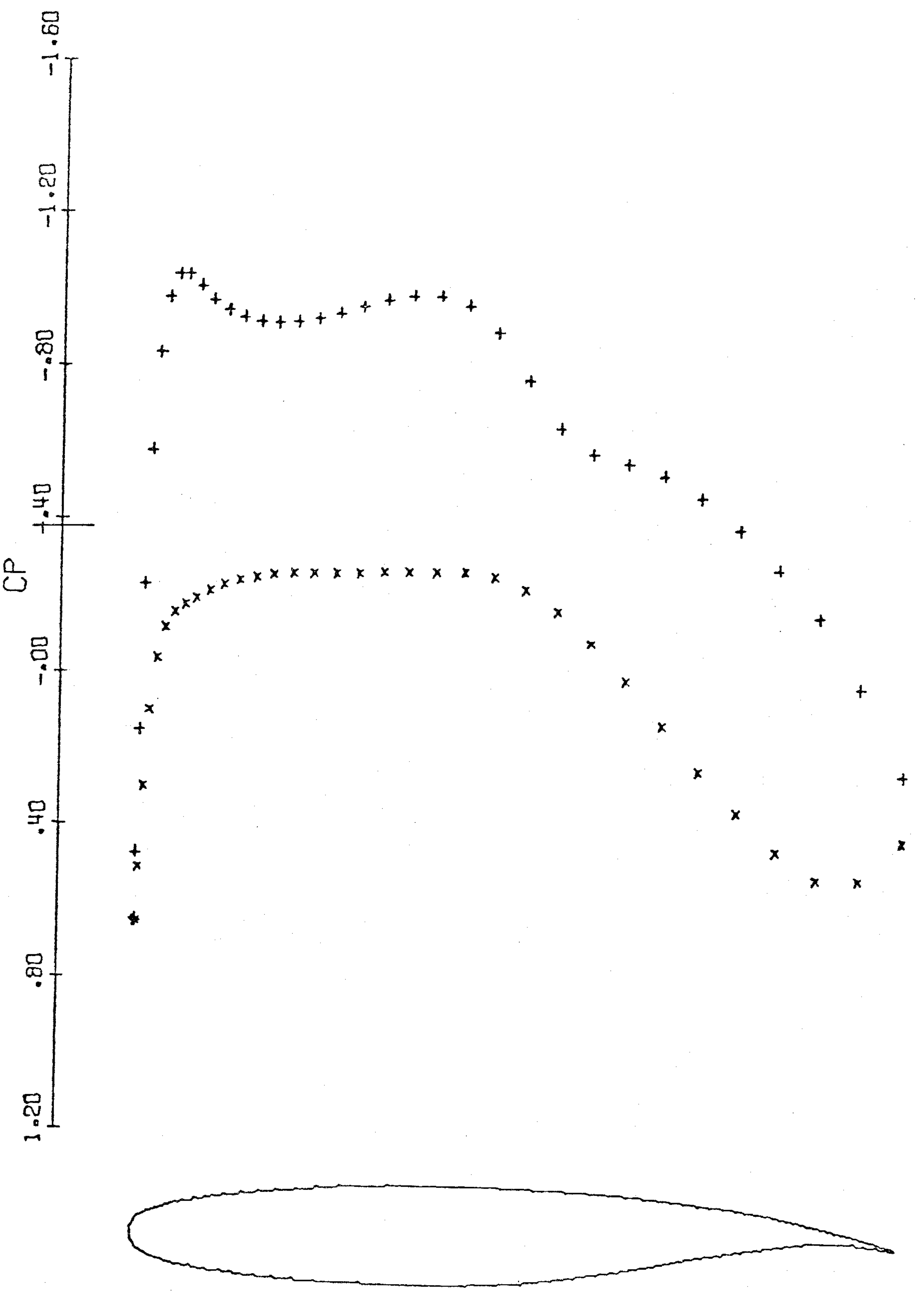
FIGURE 5(E)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)

MACH	.819	YAW	0.000	ALPHA	0.000
Z	7.20	CL	.6263	CD	-.0022

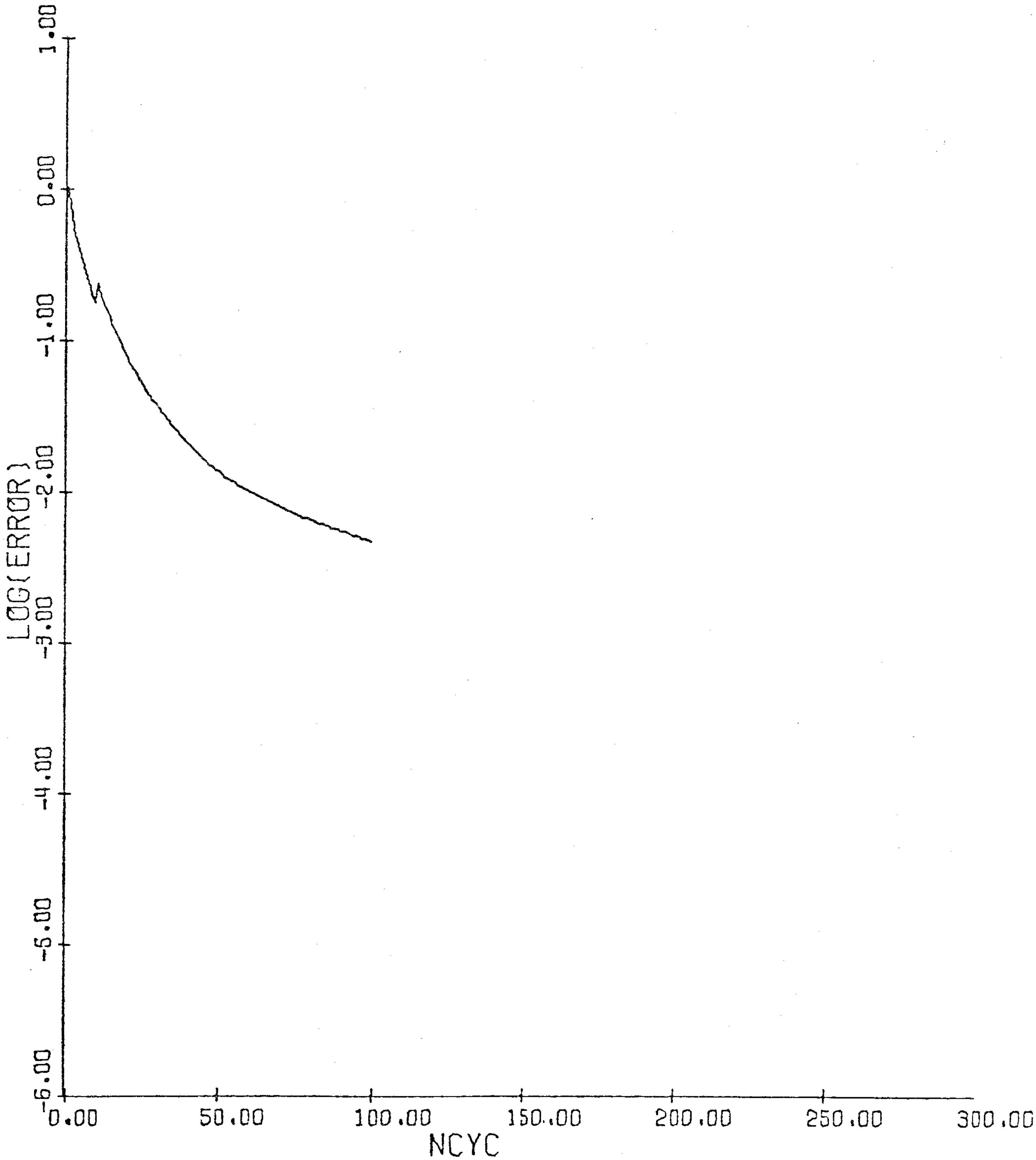
FIGURE 5(F)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)

MACH	.819	YAW	0.000	ALPHA	0.000
Z	9.60	CL	.6475	CD	-.0164

FIGURE 5(G)



DOUGLAS WING W2 (EXTENDED TO CENTER LINE)

MACH	.819	YAW	0.000	ALPHA	0.000
RES1	-.130E-01	RES2	-.599E-04		
WORK1	0.00	WORK2	99.00	RATE	.9471
GRID	144 X 24 X 32				

FIGURE 6

INPUT DESCRIPTION

Read Order	Number Cards		Description and Comments
1	1	TITLE.	- Descriptive title of case or sequence; Format (8A10) Appears on Varian Plots and beginning of output.
2	1	DESC.	- Description for card in Read Order 3 Format (8A10).
3	1	FNX, FNY,	FNZ, FPLOT, XSCALE, PSCALE, FCONT Format (8E10.7)
			Note: A number of quantities are read in as floating-point numbers and converted to integers within the program.
		FNX.	- Number of computational grid points in "chordwise direction" from downstream infinity, around the leading edge and back to downstream infinity on coarsest mesh. Maximum is 96 (192 with no grid halving).
		FNY.	- Number of computational grid points in "normal direction" from airfoil surface to infinity on coarsest mesh. Maximum is 12 (24 with no grid halving).
		FNZ.	- Number of computational grid points in "spanwise direction" from infinity, across the wing span and to infinity on coarsest mesh. Maximum is 16 (32 with no grid halving).
		FPLOT.	- Plot trigger. Selects type of plot for chordwise surface pressure coefficients. FPLOT = 0. Printer plots, one at each spanwise grid plane section with CP versus the computational grid chordwise variable.

Read Order	Number Cards		Description and Comments
			FPLOT = 1. Varian plots (from THREED). These are superimposed plots, with all span sections shown on two figures, an upper surface and a lower surface plot of CP versus physical space chord-wise variable.
			FPLOT = 2. Varian plots (from THREED) as above plus section plots (from GRAPH). These latter plots, one per section, give upper and lower surface CP versus physical space chordwise variable.
			Defaults to zero
		XSCALE.	- Scale of abcissa in pressure plots at each wing station: XSCALE > 0., Length of section airfoil chord (in inches); XSCALE = 0., 5-inch section chord; XSCALE < 0., Length of maximum section airfoil chord will be |XSCALE| inches on plot. All other section plots will be scaled according to wing planform.
		PSCALE.	- Scale of ordinate in pressure plots, C_p on plot: PSCALE = 0., -.4 per inch. PSCALE ≠ 0., - |PSCALE| per inch.
		FCONT.	- Program starting/continuing trigger. FCONT = 0., Calculation begins at iteration zero. FCONT = 1., Calculation is to be continued from a previous one.
4	1	DESC.	- Description for card in Read Order 5 Format (8A10).
5	1 card for each computational grid. Maximum essentially 2, dimensioned 3.	FIT, COVO, P10, P20, P30, BETAO, STRIPO, FHALF	Format (8E10.7)
		FIT.	- Maximum number of iterations on this gird, called MIT in program.
		COVO.	- Convergence criterion on the maximum change in reduced velocity potential (G) from one iteration cycle to the next on this grid.

Read Order	Number Cards	Description and Comments

P10. - Subsonic point relaxation factor on this grid; must be <2.. Typically 1.6 on coarse grid.

P20. - Supersonic point relaxation factor; must be ≤ 1.. Should use 1.0 for stability on all grids.

P30. - Circulation relaxation factor. May be > 1.0 .

BETAO. - Stabilization factor used at supersonic points in finite difference operator if BETAO > 0. Most needed when M_∞ >1., many cases operate satisfactorily with BETAO = 0. Convergence is slowed but stability enhanced when BETAO >0.

STRIPO. - Line relaxation control. Computational X-Y planes are relaxed by horizontal lines (YSWEEP) in central strip, vertical lines (XSWEEP) in outer strips. STRIPO specifies the fraction of computational plane included in central strip: 0. ≤ STRIPO ≤ 1., where STRIPO=1. gives all horizontal line relaxation.

FHALF. - Grid halving trigger.
|FHALF| ≥ 1. read another card (Read Order 5 format) containing computational parameters to be used on grid with mesh size halved in all directions.
|FHALF| < 1. must appear on finest grid card (last one read). Calculation proceeds automatically through the sequence of computational grids.

6 1 DESC. - Description for card in Read Order 7 Format (8A10).

7 1 FMACH, YA, AL, CDO
Format (8E10.7)

FMACH. - Freestream Mach number.

YA. - Yaw angle (in degrees).

AL. - Angle of attack (in degrees) measured in plane containing freestream direction.

CDO. - Drag coefficient due to skin friction (CD FRICTION on output). This input number is added to the drag coefficient obtained by integrating the surface pressures (CD FORM on output).

Read Order	Number Cards	Description and Comments

Read Orders 8 through 19 are used to specify the wing geometry (in physical space, of course). One can define the wing at up to 11 span stations. A set of airfoil coordinates <u>must</u> be read in at the first station. It need not be read in at other stations, if one is changing only combinations of the following three airfoil section parameters: chord, thickness ratio or angle of attack (twist). The wing shape at intermediate span positions (i.e., the computational grid planes for example) is obtained by linear interpolation in the spanwise direction in the physical space.

Read Orders 8 and 9 are read only once: 10 and 11 are read FNC (see 9) times; 12 through 17 (19 if non-symmetric airfoil section) must be read at first section and may be required at other sections, depending on the wing geometry.

8 | 1 | DESC. - Description for card in Read Order 9 Format (8A10).

9 | 1 | ZSYM, FNC, SWEEP 1, SWEEP 2, SWEEP, DIHED 1, DIHED 2, DIHED

ZSYM. - Wing planform symmetry trigger:
ZSYM = 0, yawed wing
ZSYM = 1, swept wing

FNC.- - <u>Must be ≥ 3</u>. The leading edge of the wing in physical space is fit with a cubic spline. Data at three span stations are required (as minimum) as well as the six angles which follow. If the wing leading edge has a slope discontinuity, three stations should be used fairly close to it.

SWEEP 1. - Sweep angle of wing leading edge at root section (in degrees).

SWEEP 2. - Sweep angle of wing leading edge at tip section (in degrees).

SWEEP. - Sweep angle of spanwise grid lines at ∞ (off tip of wing) (in degrees).

DIHED 1. - Dihedral angle of wing leading edge at root section (in degrees).

DIHED 2. - Dihedral angle of wing leading edge at tip section (in degrees).

DIHED. - Dihedral angle of spanwise grid lines at ∞ (off tip of wing) (in degrees).

Read Order	Number Cards	Description and Comment
10	1	DESC. - Description for cards in Read Order 11 Format (8A10).
11	1	ZS(K), XL, YL, CHORD, THICK, AL, FSEC Format (8E10.7)
		ZS(K). - Spanwise coordinate of the wing section being specified. It is in the same units as CHORD. These stations are ordered from tip-to-tip, in ascending algebraic order of ZS(K) for yawed wing and root-to-tip for swept wing.
		XL. - X coordinate of section leading edge in physical space (controls sweep).
		YL. - Y coordinate of section leading edge in physical space (controls dihedral).
		CHORD. - Section chord length. The chord of the airfoil coordinates to be read in (or already read in at the prior station) will be scaled to this value.
		THICK. - Section thickness ratio relative to that of the airfoil coordinates to be read in (or already read in at the prior station). Note, this is a ratio of thickness/chord ratios. The thickness of the airfoil coordinates will be scaled with this value.
		AL. - Section angle of attack or twist (in degrees). Airfoil coordinates will be rotated through this angle about LE.
		FSEC. - Section airfoil coordinate trigger. FSEC = 0. Do not read airfoil coordinates. Last set of airfoil coordinates read will be used at this section. They may be scaled by any combination of CHORD, THICK, or AL read above. Skip Read Orders 12 through 19 for this section.

Read Order	Number Cards			Description and Comments
				FSEC = 1. Read a new set of airfoil coordinates which will be used at this station and perhaps at other stations. They may be scaled by any combination of CHORD, THICK, or AL read above for this section. At first station (K = 1) FSEC is ignored; one <u>must</u> supply Read Orders 12 through 17.
12	1	DESC.	-	Description for cards in Read Order 13 Format (8A10).
13	1	FSYM, FNU, FNL		Format (8E10.7)
		FSYM.	-	Airfoil symmetry trigger. FSYM $\geq$ 1. Symmetric airfoil. Read in only upper surface airfoil coordinates, ordered leading edge to trailing edge. FSYM <1. Non-symmetric airfoil. Read in upper and lower surface airfoil coordinates, respectively, each set ordered leading edge to trailing edge. Note that leading-edge points are included in both sets.
		FNU.	-	Number of coordinates read in for upper surface of airfoil.
		FNL.	-	Number of coordinates read in for lower surface of airfoil.
14	1	DESC.	-	Description for cards in Read Order 15 Format (8A10).
15	1	TRL, SLT, XSING, YSING		Format (8E10.7)
		TRL.	-	Included angle of trailing edge of airfoil (in degrees). For blunt TE it is upper slope angle minus lower slope angle.
		SLT.	-	Slope of airfoil mean camber line at trailing edge.

Read Order	Number Cards		Description and Comments
		XSING. -	X coordinate of the origin of the mapping referenced to the airfoil leading edge. Recommend approximately X(LE) + 1/2 leading edge radius where R_{LE} is in the same units as XP(I) read below.
		YSING. -	Y coordinate of the origin of the mapping referenced to the airfoil leading edge. Recommend approximately Y(LE).
16	1	DESC. -	Description for cards in Read Order 17 Format (8A10).
17	FNU	XP(I), YP(I)	Format (8E10.7)
		XP(I). -	X coordinate of airfoil upper surface, ordered leading edge to trailing edge.
		YP(I). -	Y coordinate of airfoil upper surface, ordered leading edge to trailing edge. Note that there is only one pair of coordinates per card.

If airfoil section is not symmetric (FSYM<1.) the airfoil lower surface coordinates must be read here. For symmetric airfoil (FSYM≥1.), skip the two Read Orders 18 and 19.

Read Order	Number Cards		Description and Comments
18	1	DESC. -	Description for cards in Read Order 19 Format (8A10).
19	FNL	VAL, DUM -	Format (8E10.7)
		VAL. -	X coordinate of airfoil lower surface, ordered leading edge to trailing edge.
		DUM. -	Y coordinate of airfoil lower surface, ordered leading edge to trailing edge. Note that there is only one pair of coordinates per card.

Read Orders 10 through 19 complete the input for one span station. As indicated above Read Order 8, at least Read Orders 10 and 11 must be repeated for the remaining FNC-1 sections when FNC ≥ 3.

Read Order	Number Cards		Description and Comments

The program terminates by reading the first three Read Orders with FNX<1.; that is, last three cards for a normal stop should be:

1	1	TITLE.	- End of Calculation
2	1	DESC.	- Description for card in Read Order 3
3	1		0. . . . .

REFERENCES

1. Jameson, Antony: Transonic Flow Calculations. Presented at the Lecture Series on Computational Fluid Dynamics, March 15-19, 1976, von Karman Institute, Rhode-St-Genese, Belgium. (Lecture notes to be published.)

2. Newman, P. A.; and Davis, R. M.: Input Description for Jameson's Three-Dimensional Transonic Airfoil Analysis Program. NASA TMX-71919, 1974.