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CFD02-FORTRAN PROGRAM FOR ACCURATE ANALYSIS OF STEADY COMPRESSIBLE AIRFOIL AND CASCADE FLOWS USING HIGH ORDER SURFACE PANEL METHOD

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ABSTRACT

The user-oriented computer program, CFD02, is applicable to aerodynamic analysis of steady, inviscid, irrotational flows about isolated two-dimesional airfoils and cascades of airfoils. The program has option to use either first or second order vortex panel method to solve Laplace's equation which governs potential flow field around any arbitrary airfoil shape or cascade configurations. Thin airfoil theory Kutta-Zhukowski condition is used to determine correct circulation around the airfoil.

INTRODUCTION

Panel methods, which are a sub-family of boundary element methods, have been widely used for aerodynamic analysis and design of two and three dimensional configurations because of computational efficiency, versatility, and accuracy [1]. Boundary element formulation can treat not only external flow problems but also internal flows, such as the flow field through turbomachinery cascades. However, the accuracy of panel methods strongly depends on the choice of surface singularity [2] and the treatment of Kutta-Zhukowski condition [3]. The order of expansion for panel configurations and unknown strength of singularities along the panels is another important factor in obtaining highly accurate numerical results [4].

The present program uses surface vortex panel method which is equivalent to doublet panel method. It utilizes linear finite element expansion for unknown vortex strength along each panel. Geometrically curved panels are used to interpolate the surface configuration of an airfoil and the interpolation is obtained using parabolic curvature fitting. Hence, the computational results for both isolated airfoil and cascade of airfoils give second order accuracy. Kutta-Zhukowski condition used in thin airfoil theory is explicitly enforced at the trailing edge of the airfoil. In order to extend the panel method to compressible flow analysis, either field source term discretization, which represents nonlinear compressibility effect [5], or algebraic compressibility corrections [6], which are slightly less accurate but more computationaly efficent, must be introduced. The present program includes three well known algebraic compressibility corrections as options.

ANALYSIS

The continuity equation for a steady incompressible irrotational flow can be expressed as Laplace's equation,

$$\nabla^2\phi = 0 \tag{1}$$

Using Green's function formulation, an integral equation is obtained as [7]

$$\int \gamma(s)\,\frac{\partial\phi}{\partial n}\,ds = \vec{n}\cdot\vec{V}_\infty \tag{2}$$

where $\gamma(s)$ is the unknown vortex strength distribution. It is discretized using linear finite element expansion. Here, ϕ is velocity potential function. The potential with unit strength can be expressed as

$$\phi = \frac{1}{2\pi}\,\theta \tag{3}$$

For a cascade of point vortices, the potential becomes

$$\phi = -\frac{1}{2\pi}\tan^{-1}[\coth\frac{\pi}{t}(x - \xi)\ \tan\frac{\pi}{t}(y - \eta)] \tag{4}$$

where t is gap between the individual vortices in a cascade. Resulting system of linear algebraic equations is

$$\sum_{j=1}^{N} A_{ij}\gamma_j = B_i \qquad 1 \leq i \leq N-1 \tag{5}$$

Explicit enforcement of the Kutta-Zhukowski condition means that

$$\gamma_1 = \gamma_N = 0 \tag{6}$$

where subscripts 1 and N denote lower and upper trailing edge, respectively (Figure 1a). Finally the set of equations becomes

$$\sum_{j=2}^{N-1} A_{ij}\gamma_j = B_i \qquad 1 \leq i \leq N-1 \tag{7}$$

This system was solved using least squares method since the system is overdetermined. Three well known compressibility corrections were included in this program. They are:

Prandtl-Glauert compressibility correction

$$C_{p_c} = \frac{C_p}{\sqrt{1 - M_\infty^2}} \tag{8}$$

where M_∞ is the free stream Mach number at upstream infinity and C_p is the pressure coefficient corresponding to the specified free stream Mach number.

Karman-Tsien compressibility correction

$$C_{p_c} = \frac{C_p}{\sqrt{1 - M_\infty^2} + \{M_\infty^2/(1 + \sqrt{1 - M_\infty^2})\}\ C_p/2} \tag{9}$$

Laitone's compressibility correction

$$C_{p_c} = \frac{C_p}{\sqrt{1 - M_\infty^2} + \{M_\infty^2\ \left(1 + (\frac{\gamma-1}{2})\right)\ M_\infty^2/2\ \sqrt{1 - M_\infty^2}\}\ C_p} \tag{10}$$

where γ is the ratio of the specific heats of the gas.

GENERAL DESCRIPTION OF THE COMPUTER PROGRAM

Computer program CFD02 is capable of analysing both incompressible and compressibles flow around an isolated airfoil and a cascade of airfoils. CFD02 consists of the master program and subroutines. The main program includes READ statements of input data and generation of new node points. Therefore, input airfoil shape can be arbitrarily shaped, sized, and positioned and the program translates, rotates, and scales the input geometry according to the specified input parameters.

All subroutines are listed in Table 1.

TABLE 1. List of Subroutines

SPLIF	Spline fitting of input points.
INTPL	Interpolation to create new node points.
RESULT	Calculation of aerodynamic parameters: C_p, C_L, C_D, C_M.
GAUSS	Matrix inversion using Gauss elimination.
MTRX4	Construction of set of linear equations.
NEWCOOR	Parabolic curve fitting for surface panels.

Subroutines SPLIF and INTPL are used to interpolate the input surface coordinates and determine new surface coordinates using cubic spline fitting and interpolation. Subroutine COEVX28 computes influence coefficients using numerical integration. Subroutine MTRX4 constructs coefficient matrix A_{ij} using the results obtained by subroutine COEVX28 and known vector B_i constituting right hand side of equation (7). Then, subroutine GAUSS solves the system of equation (7). Subroutine RESULT computes tangential velocity and pressure coefficient $C_p=(P-P_\infty)/(1/2\rho U_\infty)$ at each control point using the solution of subroutine GAUSS. Lift coefficient C_L, drag coefficient C_D, moment coefficient C_M, and pressure coefficient C_p for compressible flow using three compressibility correction mentioned before are also computed in subroutine RESULT.

INPUT DATA DESCRIPTION

The entire input data set is read in an unformated form. The first line (see example in Figure 2) contains an arbitrary text with up to 80 alphanumeric characters describing the airfoil or test case. The following input lines contain :

NPANELS Number of panels to be generated (max. 80). This must be an even integer.

INPNTS Total number of input surface points of which X, Y coordinates are given as input data.

CLUST Coefficient of clustering function used to generate new surface points. Maximum value is $1/2\Pi$ and suggested value is about 0.10 - 0.14.

CHORD Actual maximum chord length of the airfoil.

GTC	Gap to chord ratio. This is an important parameter for cascade. For free airfoil analysis, give it an arbitrary large value, say, GTC = 100.0.
ASTG	Stagger angle (in degrees) between reference axis and x axis. ASTG > 0.0 means that the leading edge is down. ASTG < 0.0 means that the leading edge is up.
TC	Ratio of actual maximum relative thickness and maximum relative thickness of the input airfoil. Actual airfoil contour can be made either thicker or thinner using this option.
EMFREE	Free stream Mach number at upstream infinity.
AL1	Angle of attack in degrees for isolated airfoils (Figure.1a). For cascades, AL1 is free stream angle obtained as AL1 = arctan ((tan (ALFIN) + tan (ALFOUT))/2.0) where ALFIN is inlet free stream angle at upstream infinity and ALFOUT is outlet free stream angle at downstream infinity (Figure.1b).
GAS	Ratio of specific heats of the gas. GAS = 1.667 for monoatomic gas GAS = 1.4 for diatomic gas GAS = 1.0 - 1.4 for polyatomic gas
ALFIN	Inlet free stream angle in degrees (Figure 1b). ALFIN is arbitrary for isolated airfoils.
INDX1	Index for the trailing edge geometry INDX1 = 1 Round trailing edge

INDX1 = 2 Cusped trailing edge

INDX2 Indes for the leading edge geometry

INDX2 = 1 Symmetric airfoil or cascade leading edge curvature correction

INDX2 = 2 Asymmetric airfoil or cascade leading edge curvature correction

INDX3 Index for panel configuration.

INDX3 = 1 straight panel

INDX3 = 2 curved panel

INDX4 Index for compressibility correction

INDX4 = 0 no correction is performed

INDX4 = 1 Prandtl-Glauert compressibility correction

INDX4 = 2 Karman-Tsien compressibility correction

INDX4 = 3 Laitone compressibility correction

XS1,YS1 X and Y coordinates of the leading edge reference point

XS2,YS2 X and Y coordinates of the trailing edge reference point

XB,YB X and Y coordinates of input airfoil surface points. Total number of input data points must be INPNTS.

OUTPUT DATA DESCRIPTION

Output starts with an echo printout of the input data followed by the computational results described below (see example Figure.3).

X,Y Coordinates of each control point.

CP Incompressible or compressible pressure coefficient at each control point.

CL Lift coefficient.

CD Drag coefficient.

CMLE Moment coefficient about leading edge.

CX,CY X and Y direction force coefficients.

CMAC Moment coefficient about quarter chord.

XCA,YCA Coordinates at which pressure coefficient C_p is printed. Surface pressure coefficient C_p is printer-plotted on the output paper. Results are permanently saved on unit 13.

LIMITATIONS AND APPLICABILITY OF COMPUTER PROGRAM CFD02

Computer program CFD02 is directly applicable to subsonic irrotational two-dimensional steady inviscid flow analysis over a given airfoil or a cascade of airfoils. There is almost no limitation about airfoil geometry. However, Kutta-Zhukowski condition is always enforced at the trailing edge. Up to free stream Mach number 0.5 the calculation results show high accuracy for relatively small angles of attack [7]. Whatever angle of attack or free stream Mach number is specified, no separation can be simulated. Therefore, for high angles of attack unrealistic C_p values will be observed.

REFERENCES

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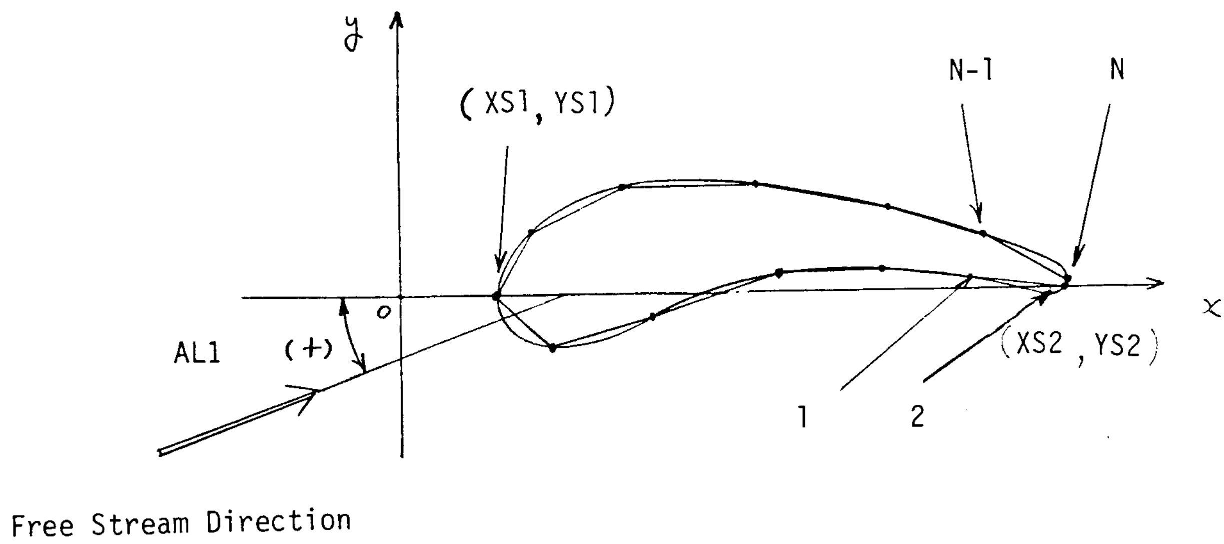

Figure 1a. Notations for input data (Airfoil)

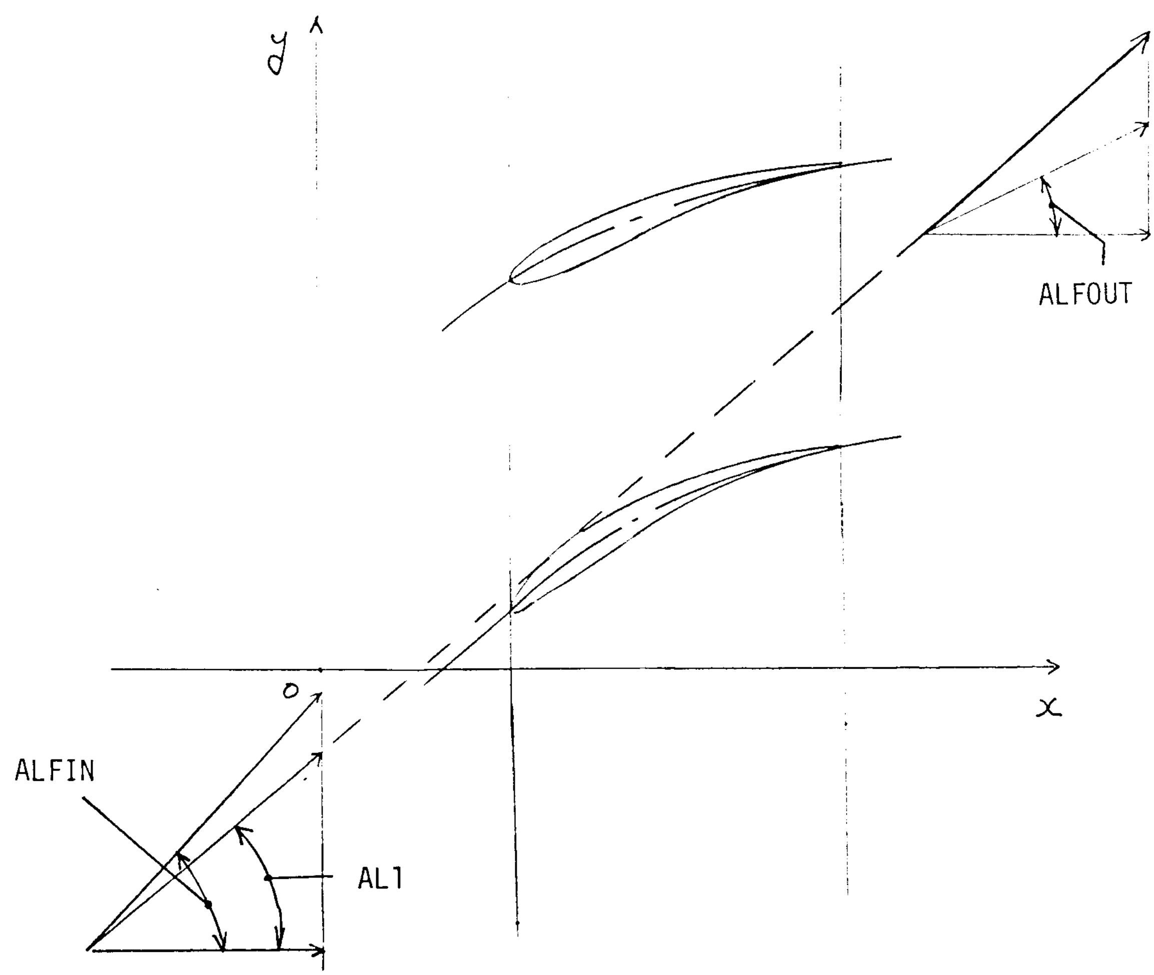

Figure.1b Notation for Input Data (Cascade of Airfoils)

Figure 2.

Example Input Data

GOSTELOW CASCADE AIRFOIL

NPANELS	INPNTS	CLUST	CHORD	GTC	ASTG	TC	EMFREE	AL1	GAS	ABFIN
32	86	0.14	1.0	0.9901	37.5	1.0	0.001	44.0	1.4	53.5

XS1	YS1	XS2	YS2
0.000	0.0000	0.999991	0.0

INDX4	INDX5	INDX6	INDX7	INDX8
1	2	0	1	1

X	Y
0.999991	0.000000
0.998595	0.000385
0.988072	0.003036
0.979032	0.005228
0.967645	0.007734
0.953763	0.010667
0.937419	0.013788
0.918575	0.017052
0.897176	0.020363
0.873147	0.023617
0.846384	0.026703
0.816756	0.029505
0.784036	0.031892
0.748138	0.033724
0.708536	0.034839
0.691659	0.035043
0.664954	0.035048
0.616568	0.034120
0.562254	0.031742
0.499957	0.027440
0.425136	0.020330
0.321930	0.007734
0.270853	0.000795
0.247212	-0.002507
0.232032	-0.004626
0.178493	-0.011872
0.165366	-0.013534
0.146439	-0.015791
0.145132	-0.015940
0.110932	-0.019354
0.057532	-0.021491
0.030894	-0.019222
0.015190	-0.015098
0.005936	-0.009925
0.001248	-0.004101
0.000063	-0.000393
-0.000141	0.002144
0.001131	0.008657
0.004615	0.015335
0.009997	0.022098
0.017058	0.028837
0.025639	0.035653
0.035627	0.042358
0.046942	0.048966
0.059532	0.055447
0.073366	0.061772
0.088437	0.067915
0.104755	0.073846
0.122350	0.079540
0.141277	0.084966
0.161613	0.090091
0.183466	0.094877
0.184036	0.094993

0.245855	0.105050
0.259913	0.106714
0.274603	0.103231
0.289989	0.109585
0.306148	0.110761
0.323177	0.111738
0.341198	0.112491
0.360367	0.112988
0.380691	0.113136
0.403055	0.113028
0.427269	0.112434
0.454158	0.111280
0.484771	0.109361
0.521145	0.106282
0.563504	0.101040
0.594192	0.097637
0.650079	0.088934
0.710527	0.077591
0.763025	0.065180
0.786059	0.060733
0.826604	0.050519
0.855912	0.042661
0.879212	0.036150
0.929087	0.021530
0.941336	0.017826
0.951990	0.014578
0.961258	0.011740
0.969296	0.009275
0.987109	0.003833
0.998633	0.000378
0.999991	0.000000

Figure 3 Example Output Data

GOSTELOW CASCADE AIRFOIL

0.00000	0.79092	0.56173	0.00000	0.79466	0.56173	1
0.00406	0.77963	0.57717	0.00810	0.78718	0.59116	2
.01638	0.75762	0.61375	0.02462	0.77208	0.62611	3
0.04072	0.72004	0.63262	0.05670	0.74317	0.63557	4
0.08303	0.66501	0.64118	0.10912	0.69691	0.64454	5
0.14654	0.59397	0.64569	0.18354	0.63311	0.64438	6
0.23123	0.51118	0.63992	0.27829	0.55483	0.63273	7
0.33339	0.42260	0.62105	0.38861	0.46753	0.60653	8
0.44856	0.33450	0.58716	0.50735	0.37767	0.56446	9
0.56748	0.25232	0.53695	0.62610	0.29134	0.50549	10
0.68219	0.17994	0.47095	0.73643	0.21331	0.43320	11
0.78491	0.11953	0.39509	0.83120	0.14657	0.35607	12
0.86964	0.07192	0.32824	0.90563	0.09250	0.31376	13
0.93310	0.03707	0.31899	0.95804	0.05135	0.36505	14
0.97518	0.01461	0.43313	0.99005	0.02279	0.67827	15
1.00639	0.00322	0.99999	1.00639	0.00644	0.99999	16
1.01835	-0.00133	0.11754	1.01440	0.00000	0.39463	17
1.03213	-0.00362	-0.45738	1.02371	-0.00365	-0.13688	18
1.05754	0.00108	-0.72497	1.04143	-0.00359	-0.64301	19
1.10115	0.01812	-0.80081	1.07489	0.00575	-0.75749	20
1.16620	0.05274	-0.76728	1.12389	0.03049	-0.80551	21
1.25257	0.10791	-0.65527	1.20511	0.07499	-0.71863	22
1.35731	0.13337	-0.48628	1.30138	0.14033	-0.57934	23
1.47411	0.27520	-0.29222	1.41436	0.22591	-0.39026	24
1.59522	0.37640	-0.10622	1.53535	0.32449	-0.19567	25
1.71211	0.47835	0.05353	1.65636	0.42831	-0.02049	26
1.81691	0.57252	0.18509	1.76888	0.52838	0.12536	27
1.90358	0.65199	0.29227	1.86569	0.61665	0.24442	28
1.96835	0.71258	0.37784	1.94197	0.68733	0.34321	29
2.01272	0.75356	0.44345	1.99603	0.73784	0.41355	30
2.03843	0.77762	0.50758	2.02955	0.76928	0.48391	31
2.05201	0.79031	0.54628	2.04736	0.78596	0.53226	32

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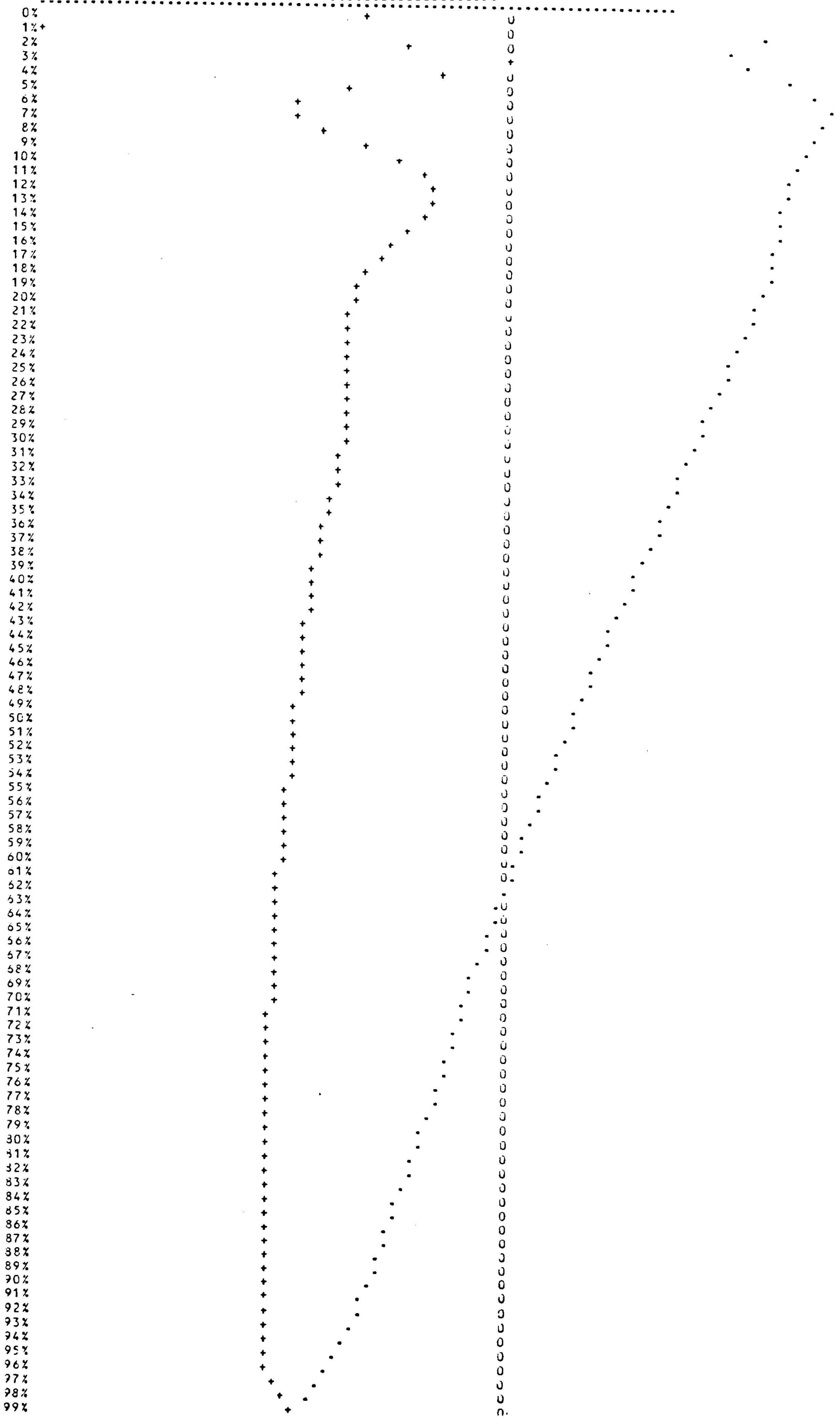

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