FOILGEN

This program is used for airfoil geometry generation. For airfoils with analytically defined ordinates, described in App. A. Geometry for Aerodynamics, the program produces airfoil definition datasets in the format required for airfoil analysis codes. The ordinates are output in a standard of two columns containing the *x*/c and *y/c* coordinates as shown below. Options include NACA 4-digit, 4-digit modified and 5-digit airfoil thickness and camber shapes. In addition, the NACA 6 and 6A camber lines are available. The user can combine any combination of thickness forms and camber lines available within these shapes. This provides a wide range of airfoil geometries. The program runs interactively, and a sample terminal session is provided here to illustrate its use. Since typical airfoil definitions include a small finite thickness at the trailing edge, the user must make sure the particular airfoil analysis code can handle a finite thickness trailing edge.

From the terminal session:

```
NACA Airfoil Ordinate Generation

    W.H. Mason, March 15, 1992

Thickness Distribution Options:

      1 - NACA 4 Digit Series
      2 - NACA Modified 4 Digit Series

Select 1 or 2 :2

Input Max Thickness,  T/C =.18

X/C Position of Max Thickness =.4

Input leading edge parameter:
Choose values from 0 to 9 -
      (6 is the 4 Series value)  7

Leading Edge Radius, rle/C = 0.04859

Trailing Edge Angle is 31.60 degrees
     [this is the TOTAL included angle]

Camber Distribution Options:

      1 - NACA 4 Digit Series
      2 - NACA 5 Digit Series
      3 - NACA 6 & 6A  Series

Select 1,2 or 3: 3

Design Lift Coefficient = .2

Input X/C for constant loading, A = .8

6A-series camber line ? (Y/N):y

Choose output option :

      1 - Point by point
      2 - Distribution

      Select 1 or 2:2
```

Select type of distribution:

1 - Even Spacing
2 - Full Cosine
(Concentrated at both LE & TE)
3 - Half Cosine
(Concentrated at LE)

Choose 1, 2, or 3 :1

Number of points in distribution,
(131 maximum) =21

I	X/C	YT/C	DYT/X	YC/C	DYC/C	XU/C(%)	YU/C(%)	XL/C(%)	YL/C(%)
1	0.0000	0.0000	99.9999	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
2	0.0500	0.0529	0.3774	0.0036	0.0543	4.7133	5.6407	5.2867	-4.9195
3	0.1000	0.0665	0.2020	0.0060	0.0412	9.7259	7.2420	10.2741	-6.0498
4	0.1500	0.0747	0.1343	0.0078	0.0331	14.7529	8.2487	15.2471	-6.6872
5	0.2000	0.0804	0.0969	0.0093	0.0269	19.7837	8.9702	20.2163	-7.1099
6	0.2500	0.0846	0.0706	0.0105	0.0217	24.8161	9.5086	25.1839	-7.4057
7	0.3000	0.0875	0.0478	0.0115	0.0172	29.8495	9.9018	30.1505	-7.6046
8	0.3500	0.0894	0.0249	0.0122	0.0130	34.8839	10.1599	35.1161	-7.7119
9	0.4000	0.0900	0.0000	0.0128	0.0090	39.9189	10.2786	40.0811	-7.7207
10	0.4500	0.0893	-0.0261	0.0131	0.0051	44.9543	10.2487	45.0457	-7.6202
11	0.5000	0.0874	-0.0518	0.0133	0.0012	49.9894	10.0698	50.0106	-7.4096
12	0.5500	0.0842	-0.0769	0.0133	-0.0028	55.0236	9.7439	54.9764	-7.0915
13	0.6000	0.0797	-0.1017	0.0130	-0.0071	60.0564	9.2725	59.9436	-6.6692
14	0.6500	0.0740	-0.1259	0.0125	-0.0118	65.0871	8.6561	64.9129	-6.1465
15	0.7000	0.0671	-0.1498	0.0118	-0.0172	70.1156	7.8941	69.8844	-5.5287
16	0.7500	0.0591	-0.1731	0.0108	-0.0241	75.1424	6.9835	74.8576	-4.8231
17	0.8000	0.0498	-0.1960	0.0093	-0.0361	80.1796	5.9136	79.8204	-4.0441
18	0.8500	0.0395	-0.2184	0.0072	-0.0469	85.1847	4.6629	84.8153	-3.2200
19	0.9000	0.0280	-0.2404	0.0047	-0.0480	90.1343	3.2641	89.8657	-2.3264
20	0.9500	0.0154	-0.2619	0.0023	-0.0480	95.0740	1.7693	94.9260	-1.3120
21	1.0000	0.0018	-0.2830	0.0000	0.0000	100.0000	0.1800	100.0000	-0.1800
I	X/C	YT/C	DYT/X	YC/C	DYC/C	XU/C(%)	YU/C(%)	XL/C(%)	YL/C(%)

send output to a file? (Y/N):

y

enter file name:

testout.txt

enter file title:

NACA 18% thick, xt=.4, I=7, 6A series cam, CLI = .2

Another case?

n

STOP

The disk file generated from the session shown above is:

```
NACA 18% thick, xt=.4, I=7, 6A series cam, CLI = .2
21.000000 21.000000
 Upper Surface
 0.000000  0.000000
 0.047133  0.056407
 0.097259  0.072420
 0.147529  0.082487
 0.197837  0.089702
 0.248161  0.095086
 0.298495  0.099018
 0.348839  0.101599
 0.399189  0.102786
 0.449543  0.102487
 0.499894  0.100698
 0.550236  0.097439
 0.600564  0.092725
 0.650871  0.086561
 0.701156  0.078941
 0.751424  0.069835
 0.801796  0.059136
 0.851847  0.046629
 0.901343  0.032641
 0.950740  0.017693
 1.000000  0.001800
 Lower Surface
 0.000000  0.000000
 0.052867 -0.049195
 0.102741 -0.060498
 0.152471 -0.066872
 0.202163 -0.071099
 0.251839 -0.074057
 0.301505 -0.076046
 0.351161 -0.077119
 0.400811 -0.077207
 0.450457 -0.076202
 0.500106 -0.074096
 0.549764 -0.070915
 0.599436 -0.066692
 0.649129 -0.061465
 0.698844 -0.055287
 0.748576 -0.048231
 0.798204 -0.040441
 0.848153 -0.032200
 0.898657 -0.023264
 0.949260 -0.013120
 1.000000 -0.001800
```

The user should always plot the airfoil geometry, whether obtained from foilgen or another source before undertaking an analysis. Many of the problems that arise when running a code are in fact the result of a bad input geometry. A plot of the sample case shown above is given in the figure below, clearly showing that the airfoil is not well enough defined at the leading edge. The evenly spaced airfoil coordinates are inadequate and a clustered spacing should have been selected.