Using “Airfoil” Analysis Program

Instruction Manual

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1. Introduction

The "airfoil" is an airfoil analysis computer program that is an adaptation of the original program "mcarfa" [1] and can be used to predict the aerodynamic characteristics of airfoils in subsonic, viscous flows. The computed aerodynamic characteristics include pressure distributions, lift, drag, pitch moment, transition position, and incipient separation on the airfoils.

The original program "mcarfa" combines the potential-flow solution with boundary-layer theory in an iterative manner. Hence, the interrelationship between the potential-flow solution and the boundary-layer effects is included in this program. The validation of this method was conducted by the original authors. Use of the original code has been shown that the method provides significant improvements in prediction accuracy.

The "airfoil" program was developed based on the "mcarfa" with the following modifications:

(1) Simplification of the input data procedure

(2) Ability to generate NACA airfoil geometry data

(3) Compacting output file

(4) Displaying airfoil shape and pressure distributions in a graphic manner

The purpose of the modifications is to make "airfoil" program easy to use. In fact, it is straightforward to use the program.

2. Running Procedure

The Running procedure of this program consists of three steps. First, issue the command "airfoil" to invoke the program. Then, select the option and input parameters of the flight condition that is required by the program. Finally, open the output file to get computed results using Text Editor and display the airfoil shape and pressure distributions on the airfoil under Matlab environment.

The following is an example demonstrating how to run the program:

(1) issue the command "*airfoil*"

(2) select the option and input parameters through the interface

Airfoil Definition Option:

1 -- Generate NACA airfoil

2 -- input airfoil data file

Select 1 or 2

1 <--- selected by user

Note: if 2 is selected, an input file, which defines the geometry of an airfoil, should be set up before the program is invoked. The input file format is given in section 3.

NACA Airfoil Selection:

1 -- NACA 4-Digit Airfoil

2 -- NACA Standard 5 Digit Airfoil

1 <--- selected by user

Enter NACA 4-Digit Airfoil Name:
Input Format: NACA XXXX
NACA 4412 <--- defined by user

Enter name of output file ===>
demo.out <--- defined by user

Enter Parameters:
Reference chord length = ? (ft)
1.0 <--- input parameter
angle of attack = ? (in deg)
4.0 <--- input parameter
Mach number = ? (0.01 < M < Mcr)
0.1 <--- input parameter
Reynolds number = ? (in mllions)
0.8 <--- input parameter

(3) Obtain the computed results

Open output file "demo.out" using Text Editor to get the aerodynamic characteristics of the airfoil defined by user. The contents of "demo.out" can be seen in the appendix. Issue the command "airfoil_display" under Matlab environment to display the airfoil shape and pressure distributions on the airfoil as shown in the appendix.

3. The Format of Input Data File of Airfoil

The format of input data file can be seen in "demo.inp". The following is comments on this data file:

```
naca 23012          <---   airfoil name
18.00    18.00      <--- the number of ordinates on upper and lower surface
Upper Surface
0.000000  0.000000   <--- x/c and z/c on upper surface
0.004287  0.017682
0.027749  0.037894
0.070389  0.056633
0.129502  0.069456
0.199941  0.074987
0.278454  0.075852
0.364476  0.073239
0.455094  0.067659
0.547234  0.059821
0.637770  0.050510
0.723629  0.040492
0.801893  0.030468
0.869907  0.021086
```

```
0.925358    0.012956
0.966366    0.006645
0.991541    0.002636
1.000028    0.001260
Lower Surface
0.000000    0.000000    <--- x/c and z/c on lower surface
0.012740 -0.012721
0.039779 -0.020779
0.079394 -0.026833
0.131489 -0.033035
0.197424 -0.039583
0.275808 -0.043913
0.361861 -0.045102
0.452638 -0.043529
0.545034 -0.039767
0.635893 -0.034464
0.722110 -0.028247
0.800741 -0.021690
0.869102 -0.015321
0.924859 -0.009647
0.966106 -0.005153
0.991432 -0.002260
0.999972 -0.001260
```

4. The .m file and data flies generated by the program

Four additional data files (gen_foil.dat, detail.dat, upper.dat and lower.dat) are generated during running the "airfoil" program. The "gen_foil.dat" is geometry data of airfoil generated by the program. The "datail.dat" is output file of the original program "mcarfa". The "upper.dat" and "lower.dat" are loaded by .m file "airfoil_display.m" to display the airfoil shape and pressure distributions under Matlab environment. In most case, users have no need to open these four data files.

5. Limitations

(1) Currently only NACA 4-digit Airfoil and NACA Standard 5 Digit Airfoil ordinates can be generated automatically.

(2) Mach number must be greater than 0.05 and less than criteria Mach number. If an incorrect Mach number is inputted, the program will fail to run in that particular flight condition.

6. References

(1) W.A. Stevens, S.H. Goradia, J.A.Braden, "Mathematical Model for Two dimensional Multi-Component Airfoil in Viscous Flow", NASA CR-1843, July 1971.

(2) C.L. Ladson, C.W. Brooks, " Development of a Computer Program to Obtain Ordinates for NACA 4-Digit, 4-Digit Modified, 5-Digit, and 6 Series Airfoils", NACA TM X-3284, November 1975.

Appendix - Results of the Example

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(1) Output File

The following is the contents of the output file "demo.out" for the example:

---- Output of Airfoil Program -----

TITLE -- NACA 4412

Mach number = 0.100 Reynolds Number = 0.800 million

Angle of Attack = 4.000 Ref. Chord = 1.000 feet

CL = 0.8584 CD = 0.0101 CM(C/4) = -0.0925

Transition Point:

Upper x/c = 0.31117

Lower x/c = 0.95974

Separation (Percent of Surface):

Upper = 2.272

Lower = 0.000

---- Pressure Distribution on Upper Surface ----

x/c	Zu	Cp
0.00183	0.00980	-0.28680
0.00668	0.01780	-0.84013
0.01413	0.02571	-1.10247
0.02453	0.03365	-1.20144
0.03776	0.04147	-1.22130
0.05337	0.04900	-1.21070
0.07107	0.05615	-1.19384
0.09018	0.06281	-1.17700
0.11100	0.06907	-1.16577
0.13289	0.07489	-1.17068
0.15481	0.07992	-1.17141
0.17790	0.08437	-1.15723
0.20142	0.08822	-1.14189
0.22543	0.09143	-1.12166
0.25024	0.09408	-1.09736
0.27526	0.09614	-1.07080
0.30070	0.09761	-1.03102
0.32663	0.09848	-0.98009
0.35323	0.09880	-0.93561
0.38038	0.09859	-0.89742

0.40774	0.09782	-0.85191
0.43635	0.09650	-0.80392
0.46570	0.09472	-0.75921
0.49491	0.09245	-0.70857
0.52639	0.08957	-0.65900
0.55784	0.08634	-0.61418
0.58887	0.08273	-0.56833
0.62137	0.07860	-0.52449
0.65293	0.07420	-0.47955
0.68527	0.06930	-0.43393
0.71717	0.06410	-0.38594
0.74868	0.05856	-0.33275
0.78118	0.05252	-0.27789
0.81243	0.04632	-0.22522
0.84480	0.03951	-0.18516
0.87597	0.03259	-0.14650
0.90759	0.02518	-0.09823
0.93885	0.01748	-0.03618
0.96967	0.00952	0.05493
1.00017	0.00125	0.17844

---- Pressure Distribution on Lower Surface ----

x/c	Zl	Cp
0.00000	0.00141	0.45618
0.00181	-0.00561	0.97177
0.00664	-0.01109	0.91422
0.01408	-0.01559	0.67830
0.02446	-0.01939	0.49276
0.03767	-0.02259	0.36686
0.05326	-0.02511	0.28193
0.07095	-0.02698	0.22633
0.09005	-0.02821	0.18977
0.11086	-0.02889	0.16816
0.13274	-0.02899	0.15923
0.15465	-0.02868	0.15722
0.17774	-0.02809	0.15744
0.20125	-0.02722	0.16003
0.22526	-0.02622	0.16391
0.25007	-0.02503	0.16900
0.27509	-0.02380	0.17327
0.30053	-0.02259	0.17374
0.32646	-0.02137	0.17258
0.35306	-0.02013	0.17259
0.38022	-0.01890	0.17244

0.40758	-0.01773	0.17027
0.43619	-0.01655	0.16751
0.46554	-0.01534	0.16545
0.49475	-0.01416	0.16405
0.52625	-0.01289	0.16303
0.55770	-0.01164	0.16238
0.58874	-0.01042	0.16265
0.62124	-0.00918	0.16306
0.65281	-0.00805	0.16281
0.68517	-0.00695	0.16219
0.71707	-0.00597	0.16065
0.74859	-0.00509	0.15774
0.78110	-0.00428	0.15381
0.81236	-0.00360	0.15060
0.84474	-0.00298	0.15048
0.87592	-0.00248	0.15149
0.90755	-0.00205	0.15195
0.93883	-0.00171	0.15496
0.96966	-0.00145	0.16115
0.99983	-0.00125	0.17844

(2) Airfoil Shape and Pressure Distributions