

A Quick Guide to FreeWake2015

A more detailed description of the underlying theoretical model with validation and sample applications is listed in “A Higher Order Vortex-Lattice Method with a Force-Free Wake,” by Götz Bramesfeld, Dissertation, Pennsylvania State University, August 2006.

FreeWake2020 is a modification of FreeWake2007/2008/

The program computes the inviscid loads of up to five wings (maybe more). It uses two wake models, one that is prescribed and based on Horstmann’s multiple lifting line method and one that uses a relaxed, force-free wake model. Either model has lifting lines with a spanwise circulation distribution of the shape of a second-order spline.

Table of Contents

A QUICK GUIDE TO FREEWAKE2015	1
HOW TO COMPILE	1
DIRECTORY STRUCTURE	1
INPUTS	2
INPUT FILES	2
INPUT.TXT FILE.....	2
AIRFOIL FILE	7
OUTPUT.....	9

How to compile

- g++ Source/Main_PerfCode2015.cpp -o fw
- the executable is “fw”

Directory Structure

FreeWake2015 requires several subdirectories in order to run properly. The input directories (inputs and airfoils) have to be created by user before running.

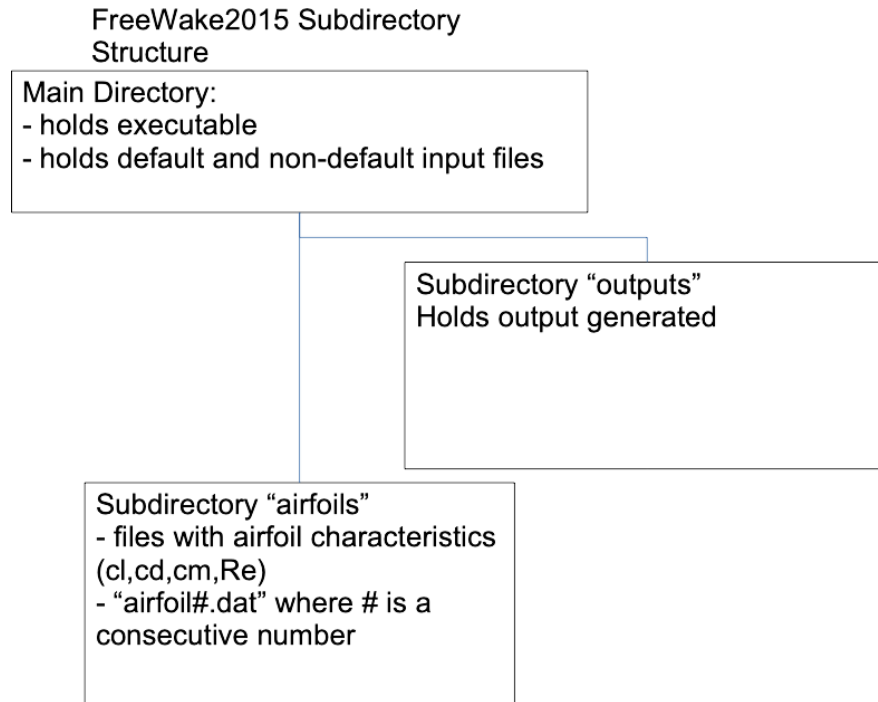


Figure 1: Subdirectory structure of FreeWake20.

Inputs

The default input file is a text file named “input.txt” that is located in the same directory as the executable. The output is written to /output.

A non-default input file is used, e.g. WingC.txt, by calling the program with the executable followed by the root of the non-default input file, e.g. ./fw WingC. The program will create the output subdirectory that has the root of the non-default input file name, e.g. /output/Wing/.

Input Files

- The lifting surface geometry and flow conditions are defined in the input file.
- Airfoil files hold aerodynamic characteristics of different airfoils (AoA, c_l , c_d , Re , c_m). The files are in the subdirectory airfoils/. The files are named airfoil#.dat, with # being consecutive numbers starting at 1. Airfoil files are only needed if viscous = 1 in input file. Note, the viscous model might be out of order (Jan. 2021)
- The following signs serve as identifiers in the input files and, thus, should be used carefully: ‘=’, ‘:’ and ‘#’. Any further spaces or comments are being ignored.

Input.txt File

- Holds configuration and general information.

- Has be in directory of executable.
- The sample input file models the wing and h-tail configuration of Fig. 2. Note that only the right wing halves are defined (symmetry flag set)

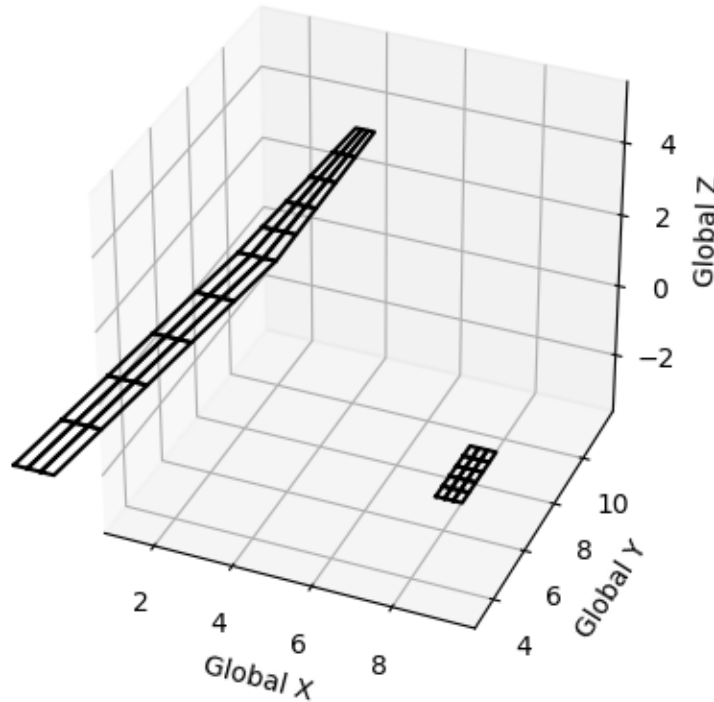


Figure 2: Right half of wing-tail configuration of sample input file.

As indicated in Fig. 2, the coordinates system is positive x in wind direction, positive y along right wing and positive z up.

In the input file, the leading-edge lines are defined for the left and right panel edges (looking in direction of flight) as is the chord and the incidence angle (positive epsilon = pitch up) of the respective edges. Thus, twist can be introduced having different epsilons on either side edge of that particular panel or by using different camber lines at left and right edge.

The wing is modeled as a flat-plate and twist is only possible through defining changing incidence angles (epsilon). Therefore, epsilon should align the panel with the zero-lift line of the particular wing section.

‘Bound.Cond.’ defines the boundary conditions at the respective edge of the panel. For example, the boundary condition of a wingtip is ‘100’, which stands for zero circulation, unknown vorticity strength and derivative thereof. The value ‘220’ is used for two or more neighboring elements (equal circulation and vorticity) and ‘010’ along the symmetry plane (unknown circulation, zero vorticity, unknown vorticity gradient).

Next, a sample input file with explanations. It is the file that was used to model the configuration in Fig. 2.

BEGINNING OF SAMPLE INPUT FILE (comments in red)

General comments, can be of any length.

Input file for FreeWake 2014

Format will not work with older versions

Input file generated by Travis's m-file, in m/N/sec

Please note that the program uses equal, number and : signs as special recognizers!

The results are written to the sub-directory output'

Simulation parameters

Relaxed wake (yes 1, no 0): relax = 0 choice of wake model
 Steady (1) or unsteady (2): aerodynamics = 1 =2 Quasi unsteady I (no apparent mass)
 Symmetrical geometry (yes 1, no 0): sym = 1 1 only right wing is modeled
 Longitudinal trim (yes 1, no 0): trim = 0 (yes means m has to be 1)

Max. number of time steps: maxtime = 20 number of timesteps

Width of each time step (sec): deltime = 0.25 Defines timestep size (with Uinf)

Convergence delta-span effic.: deltae = 0.0 (0 if only timestepping) convergence criteria

freestream flow condition

Freestream velocity (leave value 1): Uinf = 1.0
 AOA beginning, end, step size [deg]: alpha = 5.0 25.0 2.0 alpha sweep: 5 to 25 deg every 2 deg
 Sideslip angle [deg]: beta = 0.0
 Density: density = 1.
 Kinematic viscosity: nu = 1.450000e-05

Reference area: S = 25.0
 Reference span: b = 30.0
 Mean aerodynamic chord: cmac = 0.0
 Aircraft weight (N): W = 200.
 CG location (x y z): cg = 0 0 0 cg location (x,y,z) for trim
 CMo of wing: CMo = -0.1 initial pitching moment coefficient, for trim

No. of wings (max. 5): wings = 2 a t-tail is a single wing
 No. of panels: panels = 3 total number of panels. Several panels define a wing
 No. of chordwise lifting lines: m = 3 number of lifting lines across the chord, same for all panels
 No. of airfoils (max. 15): airfoils = 8 defines how many airfoil files need to be provided

general review of some parameters

Panel boundary conditions:

Symmetry line -	10
Between panels -	220
Free end -	100

Defines leading edge of wing, all measured in metres:

Definition of panels that describe the geometry. Each panel has nxm DVEs in the spanswise and chordwise direction, respectively. Each panel can have unique n values. If neighboring panels have different m-values, this may result in discontinuous circulation distribution. "overhanging" lifting lines are treated like wingtips (BC =100). Airfoil defines airfoil that is used over this panel.

First panel, panel 1 of right wing, starts here:

Panel #:1. Number of elements across span n = 5 Airfoil number = 1

Neighbouring panels (0 for none) left: 0 right: 2 panel at the right and left of panel. 0 for wingtip.

xleft yleft zleft chord epsilon Bound.Cond.

0.0 0 0.00 1.00 0.00 010 x,y,z chord, incidence angle, boundary condition (symmetry)

xright yright zright chord epsilon Bound.Cond.

1.0 0 0.000 1. 0.000 220 x,y,z chord, incidence angle, bound. cond. (neighboring element)

Panel #:2. Number of elements across span n = 5 Airfoil number = 1

Neighbouring panels (0 for none) left: 1 right: 0

xleft yleft zleft chord epsilon Bound.Cond.

1.0 10 0.000 1. 0.000 220

xright yright zright chord epsilon Bound.Cond.

2.0 15 2.000 0.5 0.000 100 x,y,z chord, incidence angle, bound. cond. (wingtip)

Right side of horizontal tail, only one panel

Panel #:3. Number of elements across span n = 5 Airfoil number = 2

Neighbouring panels (0 for none) left: 3 right: 0

xleft yleft zleft chord epsilon Bound.Cond.

10.15 .00 2.00 0.600 .00 010

xleft yleft zleft chord epsilon Bound.Cond.

10.15 2.00 2.00 0.600 .00 100

Vertical tail is as nonlifting surface modelled, that is $\text{drag} = \frac{1}{2} \rho * V^2 * S_{\text{tail}} * c_{d_profile}(\alpha=0)$

Airfoil 5 is presumably a symmetrical airfoil

%<- special identifier

Vertical tail information:

Number of panels (max 5) = 2

no.	chord	area	airfoil
1	1	1.2	5
2	1.5	2.2	5

The fuselage drag is estimated based on its washed surface area and using laminar or turbulent drag coefficients of a flat plate at zero angle of attack. Surface area of each section is estimated based on its diameter and length.

Fuselage information:

no. of sections (max 20) = 9 width of each section = .8

index of panel where transition to turbulent flow occurs = 4

no.	Diameter
1	.42
2	.68
3	.84
4	.86

5	.64
6	.48
7	.40
8	.32
9	.24

Interference for total drag

Interference drag = 1.0%

#####

END OF SAMPLE INPUT FILE

Airfoil File

- Needed when viscous =1 in geometry and configuration file.
- Holds section characteristics of airfoils. File name is airfoils/airfoil#.dat where # is a consecutive number that starts at 1.
- Number in name should correspond to camber file if camber line is used (camber = 1)
- Change in airfoil across panel possible.
- First line is a comment; five columns: alpha, cl, cd, Re, cm.
- For each Reynolds number sorted from minimum cl to maximum cl. cl has to increase, i.e. no post stall data. Reynolds number blocks sorted from lowest chord- Reynolds number to highest.
- Program uses a linear interpolation between points, thus density can be limited in low drag regions.
- Can be generated using xfoil, experiments, cfd.

BEGINNING OF SAMPLE AIRFOIL FILE

FX 67-K-170/17 0 degrees Flap Number of Rows =374					Number of rows of data
-5	0.0084	0.01632	5.00E+05	-0.107	
-4.8	0.028	0.01571	5.00E+05	-0.1067	
-4.6	0.0476	0.01518	5.00E+05	-0.1064	
.					
.					
.					
.					
8.2	1.3708	0.01158	5.00E+05	-0.104	
8.4	1.3776	0.01174	5.00E+05	-0.1016	
8.6	1.3779	0.01215	5.00E+05	-0.0983	Next Reynolds number
-5	0.0097	0.01398	7.00E+05	-0.1065	
-4.8	0.0293	0.01344	7.00E+05	-0.1062	
-4.6	0.0491	0.01299	7.00E+05	-0.1059	
.					
.					
.					
7.8	1.3455	0.00982	7.00E+05	-0.1061	
8	1.3502	0.00996	7.00E+05	-0.103	
8.25	1.3515	0.01024	7.00E+05	-0.0984	Next Reynolds number
-5	0.0086	0.01222	1.00E+06	-0.1059	
-4.8	0.0286	0.01179	1.00E+06	-0.1056	
-4.6	0.0489	0.0114	1.00E+06	-0.1055	
.					

.				
.				
7	1.3039	0.00828	1.00E+06	-0.1141
7.2	1.3195	0.00843	1.00E+06	-0.1132
7.4	1.3343	0.00855	1.00E+06	-0.1121
7.8	1.3499	0.00904	1.00E+06	-0.1073
-5	0.0002	0.00781	4.00E+06	-0.1044
-4.8	0.0224	0.0077	4.00E+06	-0.1046
.				
.				
.				
6	1.2294	0.00591	4.00E+06	-0.1202
6.2	1.2404	0.00636	4.00E+06	-0.1185
6.4	1.2463	0.00695	4.00E+06	-0.1158
6.6	1.2465	0.00762	4.00E+06	-0.112

Next Reynolds number

END OF SAMPLE AIRFOIL FILE

Output

- Several output files are being produced and written to files the ‘output’ directory. The default output directory is output/. The output files are (assuming configuration file was input.txt):
 - Performance.txt: holds a summary of an alpha sweep
 - TrimSol.txt and Performance.txt: olds trim results.
 - AOA##.txt holds lifting surface loads and circulations of ## degree angle of attack run.
 - timestep##.txt holds total flow field information (lifting surface and wake geometry) of ## degree angle of attack run.

Some general header information:

Chord: chord length of element

eta: halfspan of element

phi: sweep angle

nu: dihedral angle

epsilon: incidence angle

psi: yaw angle