

# One-Way Coupled Integral Boundary Layer

An integral boundary layer model has been added to the panel method code of the previous lab. It allows the user to compute the drag of a 2D airfoil placed in an incompressible viscous flow at prescribed Reynolds Number and angle of attack.

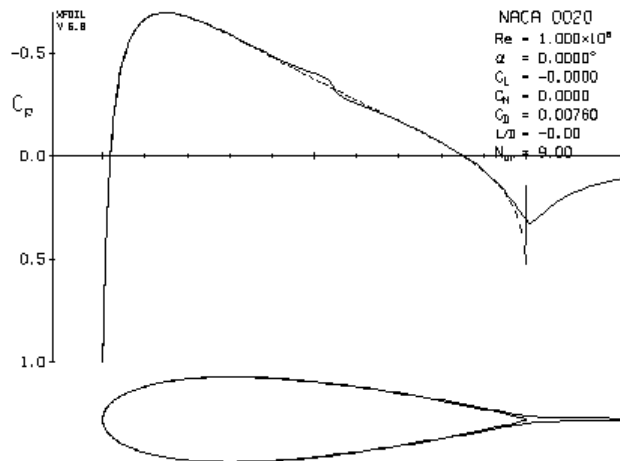
## Overview of the model

The boundary layer equations use the inviscid flow velocity provided by the panel method, but the effect of the boundary layer on the inviscid flow is not taken into account, as in Panda [1]. In more advanced developments like Xfoil [2], the coupling is performed in both ways. The model is much more complicated, and the solution is more robust and closer to experimental results.

The boundary layer model is described in [3]. It uses Thwaites' equations for the laminar part of the flow, and Head's equations for the turbulent part. Michel's criterion is used to locate transition. The drag coefficient is computed using the Squire-Young formula [4], which is a function of boundary layer parameters ( $\theta$  and  $H$ ) at the trailing edge.

## Special characteristics

The effect of the boundary layer on the inviscid flow is especially important near the trailing edge, where it reduces the adverse pressure gradient. This can be seen on the following figure, which presents results obtained using Xfoil (the solid line is the viscous  $C_p$  distribution, the dashed line is the inviscid solution).



In our case, the external velocity is not modified by the presence of the boundary layer, and we have to deal with an unrealistic very high adverse gradient at the trailing edge. The problem is that it makes the boundary layer separate very easily in the last percents of the chord length.

The solution to this problem consists in not solving the boundary layer up to the trailing edge, but up to  $C_{off}$ % of the chord length. Separation of the boundary layer which was happening in the last  $(1-C_{off})$ % of the chord is thus avoided, and the Squire-Young formula still gives good results, since it is based on a loss of momentum.

The sensitivity of the drag coefficient with respect to  $C_{off}$  will be studied in question 6 of the present lab.

## Exercises

### *1- Cylinder*

Compute and plot the boundary layer parameters ( $\theta$ ,  $\delta^*$ ,  $H$  and  $C_f$ ) on a cylinder at  $Re = 1e6$ . Use the linear vortex panel method with 50 panels. Comment on the results. What characterises laminar and turbulent separation in terms of the boundary layer parameters ?

### *2- NACA 4 digits library*

In this question, you will study the effect of the three shape parameters thickness, camber and camber location taken separately on  $c_d$ .

Use the linear vortex panel method and 100 panels.

Report  $c_d$  for 3 sets of Naca airfoils at zero angle of attack and Reynolds Number of  $1e6$  :

- \* one with variable thickness (Naca 00XX, XX varying between 04 and 26 by increment of 2),
- \* the second with variable camber (Naca X412, with X varying between 2 and 8),
- \* the third with variable camber location (Naca 4X12, with X varying between 2 and 8).

Plot the corresponding sensitivity curves  $c_d$  vs parameter. Indicate on the curves the points at which turbulent separation takes place. Draw a conclusion which sums up what you obtain on these curves.

### *3- General airfoil library*

For both 3.1 and 3.2, use the linear vortex panel method, and 100 panels.

3.1- Compute the solution for the three airfoils you selected in question 3.1 of the panel method lab. For  $Re = 1e6$ , report the drag coefficient, transition or laminar separation location and eventual turbulent separation. Comment on the differences.

3.2- Use the airfoil you selected in question 3.3 of the panel method lab. Compute the solution at  $c_l = 0$  (by finding the right angle of attack), and compare with experimental data.

### *4- Effect of the angle of attack*

Use the linear vortex panel method and 100 panels. For a Naca 0012 and a Reynolds Number of  $1e6$ , make computations of the lift and drag coefficients for  $\alpha$  varying between  $-8$  and  $+5$  with increments of 1 degree. Plot the polar curve, i.e.  $c_d$  vs  $c_l$ .

### *5- Effect of the Reynolds Number*

Choose any airfoil and compute the solution (drag coefficient, transition or laminar separation, and eventual turbulent separation) for  $Re = 1e5$ ,  $1e6$ , and  $1e7$ . Comment on the results.

## **6- Effect of cut-off distance**

In this question, you will study the influence of  $C_{off}$  on  $c_d$ . By default,  $C_{off}=0.98$ , which provides good drag results, and make turbulent separation take place for quite reasonable cases.

Use a Naca 0020 at zero alpha and at a Reynolds Number of  $1e6$ . Compute the inviscid flow with the linear vortex panel method and 100 panels.

**You will have to make the modification in the code** ( $C_{off}$  is defined in file solvebl.m at line 15).

\* Make a first try with  $C_{off}=1$ . What happens ?

\* Now, report the drag coefficient for  $C_{off} = 0.95$  to  $0.99$  by increment of  $0.01$ , and indicate the points at which turbulent separation takes place.

*Note* : the program interface gives only 4 decimals for Cd. In order to obtain a better accuracy, remove the semicolon at the end of the line where Cd is computed (file sy.m at line 3). The value of Cd will then appear in the terminal window from where you launched matlab.

Plot the curve Cd vs  $C_{off}$ . What do you observe ?

## **References**

- [1] Panda Users Guide, Desktop Aeronautics 1988. Available on [www.dektoaero.com](http://www.dektoaero.com)
- [2] M. Drela : An Analysis and Design System for Low Reynolds Number Airfoils (Xfoil). Low Reynolds Number Aerodynamics. Springer-Verlag. Lecture Notes in Eng. 54. 1989
- [3] Jack Moran : An introduction to Theoretical and Computational Aerodynamics. John Wiley and sons 1984.
- [4] H.B. Squire and A.D. Young : The calculation of the profile drag of aerofoils. R.&M. No1838. A.R.C. Technical report, 1938. London.