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PROBLEM SET 3 **Due Date: February 18th, 2005**

Solution of Inviscid-Viscous Problems Using Integral Boundary Layer Models and the Hess-Smith Panel Method

The objective of this problem set is to write a solver that uses the already-developed Hess-Smith panel method, and both Thwaites' and Head's methods for the solution of the integral momentum equation for the boundary layer to obtain viscous solutions of the *attached* flow over airfoils in incompressible flow. Notice that, for the time being, we will do this by assuming that the boundary layer *does not* influence the outer inviscid flow. This is only a two-step procedure:

1. Calculate the inviscid flow over the airfoil geometry at a specified angle of attack to obtain $V_e(x)$, $\frac{dV_e(x)}{dx}$, $\frac{dp}{dx}$ on the upper and lower surfaces of the airfoil.

2. Use the calculated $V_e(x)$, $\frac{dV_e(x)}{dx}$, $\frac{dp}{dx}$ to obtain distributions of H, θ, and δ^* along the upper and lower surfaces of the airfoil and use them to compute the coefficient of drag, C_d, of the airfoil.

To accomplish this goal, complete the following steps.

Problem 1. *Development of Thwaites' and Head's Methods for Boundary Layer Solution.* Implement in MATLAB (or the programming language of your choice) the solution procedure for both Thwaites' and Head's methods and use them to obtain *numerical* solutions of the boundary layer on a flat plate with zero pressure gradient for a Reynolds number based on the length of the plate, $Re_L = 1 \times 10^6$. Provide the following information.

- Using Thwaites' method, and assuming that the boundary layer remains laminar throughout, calculate and plot the momentum and displacement thicknesses, θ, δ^*, the shape factor, H, and the local coefficient of skin friction, c_f. Integrate c_f and provide a number for the total coefficient of skin friction, C_f for the plate. Compare all of these results to those from Blausius' solution.

- Using Head's method, and assuming that the boundary layer is tripped at the leading edge of the plate and remains turbulent throughout, calculate and plot the momentum and displacement thicknesses, θ, δ^*, the shape factor, H, and the local coefficient of skin friction, c_f. Integrate c_f and provide a number for the total coefficient of skin friction, C_f for the plate. Compare all of these results to those from Blausius' solution and to the simple power law estimates provided by Schlichting, which can be found in one of the class handouts.

Note: you can do anything you consider reasonable to deal with *undesirable* behavior at the leading edge of the flat plate.

Problem 2. *Numerical Solution of the Viscous Flow Over a NACA 3310 Airfoil with a chord Reynolds number,* $Re_c = 5 \times 10^6$. Using the program in Problem 1, and your implementation of the Hess-Smith panel method (you can also use the solution provided on the web page) construct a numerical procedure that follows the two steps explained above to construct as much of the drag polars for this airfoil as possible using Michel's transition criterion. In your discussion, address the following points:

- For the NACA 3310 airfoil at 3° angle of attack, compute and plot the following quantities for both the upper and lower surface boundary layers: θ, δ^*, H, c_f vs. distance from the leading edge stagnation point. Indicate the location of transition for both the upper and lower surfaces.
- Use the resulting program to predict the maximum lift of this airfoil, by running a c_l vs. α curve until the program predicts separation at the 80% chord location on the upper surface of the airfoil. In these calculations for $c_{l_{max}}$ you should make sure that the transition point on the upper surface is set at 1% chord. How do the values of $c_{l_{max}}$ and the corresponding angle of attack, α_{max}, compare to both experiment and the results you can obtain via XFOIL?

The results for this airfoil using XFOIL are summarized in the graph below.

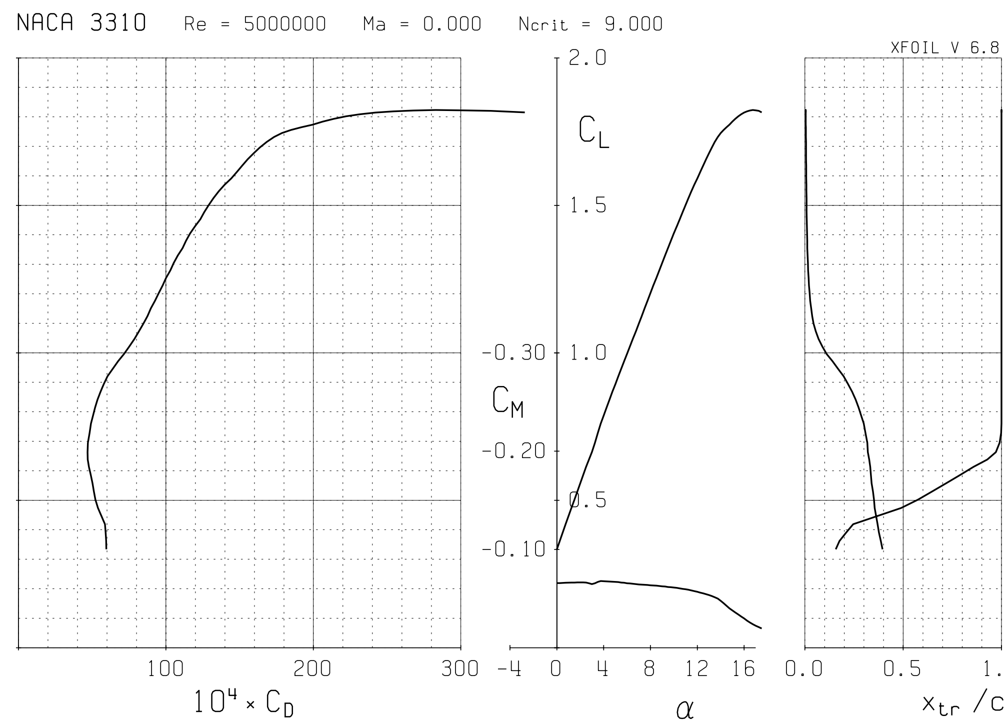

Figure 1: NACA 3310 Drag Polar