Assignment #3: Integral Boundary Layer — ME 515

due 2/26/2020 before midnight via Learning Suite — 50 possible points

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3.1 *Flat Plate*: Implement both Thwaites' and Head's methods to *numerically* solve the boundary layer flow over a flat plate with zero pressure gradient and a Reynolds number based on length of the plate of 1×10^6. The first time use Thwaites' method, assuming the boundary layer is laminar throughout, and compare your results to the analytic Blasius solution. The second time use Head's method, assuming the boundary layer is turbulent throughout, and compare your results to the empirical Schlichting formulas.

For all cases plot the following quantities along the plate: displacement thickness δ^*, momentum thickness θ, shape factor H, and local skin friction coefficient c_f. Finally, integrate c_f to compute a skin friction drag coefficient C_f.

To reiterate, you should have four separate plots and each plot should have four curves. You should also have four values for C_f (Thwaites vs Blasisus, and Heads vs Schlichting).

3.2 *Airfoil*: Use the integral momentum boundary layer equations to obtain (attached) viscous incompressible solutions for the flow over an airfoil. In general, the airfoil will be neither fully laminar nor fully turbulent so you will need to use a transition prediction method. To simplify, we will assume that the boundary layer does not significantly influence the outer inviscid flow field. Thus, you need to solve the integral boundary layer equations subject to an input inviscid solution. For this assignment we will assume only one-way coupling (no iteration needed). You can obtain an inviscid flow solution from your Hess/Smith panel method (V_e, $\frac{dV_e}{dx}$, $\frac{dp}{dx}$) or if your panel method isn't working you can obtain a pressure solution from published data or XFOIL.

Use a NACA 2412 airfoil, with a Reynolds number based on chord of 1×10^6. Complete the following:

- Plot the boundary layer properties (δ^*, θ, H, c_f) versus distance from the leading edge stagnation point. Use an angle of attack of 5°. Indicate locations of transition on both the upper and lower surfaces.
- Compute the drag coefficient (one angle of attack is fine), and compare your result to either published data (see this NACA report with aerodynamic data starting on page 131) or XFOIL (if you've never used XFOIL, it is perhaps easier to run through XFLR5).