Computer Project

This homework makes use of the vortex panel method[1] for finding the aerodynamic characteristics of airfoils. The panel method is implemented as a MATLAB function and can be downloaded from the class web site along with all the other data needed to complete this assignment. This project involves using the panel method to find the properties of two real world airfoils and then to compare these results with results obtained experimentally. The two airfoils are the NACA 2414 and the NACA 6409. The code is given at the end of this homework with an example describing it's use.

Problem 1

(a) Plot both airfoil geometries[2] in separate figures. (Note: use the MATLAB command 'axis equal' to make the scale of both axes the same.)

(b) Use the vortex panel method to determine the coefficient of lift, c_l, and moment coefficient about the aerodynamic center, c_{mac}, of both the NACA 2414 and NACA 6409 at an angle of attack $\alpha = 2.75^o$.

(c) Find the pressure distribution on both airfoils at the design angle of attack $\alpha = 3.2^o$. Plot your results with the pressure coefficient, c_p, on the y-axis and the non-dimensional chord length on the x-axis.

Problem 2

(a) Find the lift coefficient, c_l, as a function of angle of attack for both airfoils and plot your results with the experimental data[3]. Vary α between -6 and 10 degrees. (Note: you should make a separate plot for each airfoil.)

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[1] See Kuethe and Chow section 5.10 for more details.

[2] The geometry data files are given on the web site (n2414.dat, n6409.dat). The first column contains the non-dimensional chord lengths and the second column are the corresponding y ordinates. To load a file into MATLAB issue the command 'load filename.dat'.

[3] The experimental data files are given on the web site (exp2414.dat, exp6409.dat). The first column contains the angles of attach and the second column are the corresponding lift coefficients.

(b) How do the vortex panel results compare to the experimental results? Why does the vortex panel method not predict when the airfoil stalls as found in the experimental data?

(c) Find the rate of change of the lift coefficient with respect to the angle of attack, $\frac{\partial c_l}{\partial \alpha}$. Using the finite difference approximation,

$$\frac{\partial c_l}{\partial \alpha} \approx \frac{c_l(\alpha_2) - c_l(\alpha_1)}{\alpha_2 - \alpha_1}$$

Where α is in radians, $c_l(\alpha_2)$ and $c_l(\alpha_1)$ are the lift coefficients at α_2 and α_1 respectively. Compare this to the theoretically predicted value of 2π. Repeat for both airfoils.

References

[1] Abbott, I.H., Von Doenhoff, A.E.: *Theory of Wing Sections*, Dover, 1959.

[2] Kuethe, A.M., Chow, C.: *Foundations of Aerodynamics*, 5th ed., Wiley, 1998.

[3] Airfoil Geometry was obtained from: http://www.nasg.com/afdb/search-airfoil-e.phtml

[4] Experimetal Data was obtained from: http://www.aae.uiuc.edu/m-selig/pub/lsat/vol1/ and http://www.aae.uiuc.edu/m-selig/pub/lsat/vol2/.

A Vortex Panel Method: airfoil.m

```
function [cl,cmac,xp,yp,cp,V,gamma]=airfoil(XB,YB,ALPHA)
% MATLAB code based on FORTRAN code in Kuethe & Chow pages 161 - 163
%
% aifoil.m computes the airfoil lift coefficient, pitching moment coefficient about the
% aerodynamic center, as well as the pressure coefficient distribution, the velocity
% distribution, and vortex sheet strength distribution around the airfoil.  The airfoil
% contour is approximated by vortex panels of linearly varying strength.
%
% Input variables:
%      XB    row vector containing the x-coordinates of the airfoil section
%      YB    row vector containing the y-coordinates of the airfoil section
%      ALPHA angle of attack in degrees
%
% Vectors XB and YB must have the same size.  The number of panels equals the size of vectors
% XB and YB minus one. The first and last element must be the location of the trailing edge
% of the airfoil.
%
% Output variables:
%      cl    section lift coefficient
%      cmac  pitching moment coefficient about the aerodynamic center
%      xp    x location of the center point of the panels
%      yp    y location of the center point of the panels
%      cp    column vector containing the pressure coefficient at each panel. The values
%            of cp are evaluated at the center point of the panel given by (xp, yp)
%      V     column vector containing the velocity divided by the free stream
%            velocity at each panel. The values of the velocity are evaluated at the center
%            point of the panel given by (xp, yp)
%      gamma column vector containing the normalized circulation of the panels.  Note that
%            the values of gamma are the strength of the vortex sheet at points of the
%            of the airfoil given by the arrays (XB, YB)
%
% Usage:
%      1. - Typical call to obtain all the performance parameters are:
%  >XB=[1 0.933 0.75 0.5 0.25 0.067 0 0.067 0.25 0.5 0.75 0.933 1];
%  >YB=[0 -0.005 -0.017 -0.033 -0.042 -0.033 0 0.045 0.076 0.072 0.044 0.013 0];
%  >ALPHA=8;
%  >[cl,cmac,xp,yp,cp,V,gamma]=airfoil(XB,YB,ALPHA);
%  >cl,cmac
```

```
%  cl =
%    1.1792
%  cmac =
%   -0.0792
%  >cp'
%  ans =
%  Columns 1 through 7
%    0.2630    0.1969    0.2097    0.2667    0.4707    0.9929   -1.8101
%  Columns 8 through 12
%   -1.5088   -0.9334   -0.5099   -0.1688    0.1674
%
%      2. If you are interested only on cl and cmac the simpler call shown below
%  could be used.
%  >XB=[1 0.933 0.75 0.5 0.25 0.067 0 0.067 0.25 0.5 0.75 0.933 1];
%  >YB=[0 -0.005 -0.017 -0.033 -0.042 -0.033 0 0.045 0.076 0.072 0.044 0.013 0];
%  >ALPHA=8;
%  >[cl,cmac]=airfoil(XB,YB,ALPHA);
%  >cl,cmac
%  cl =
%    1.1792
%  cmac =
%   -0.0792
%      NOTE: In these examples > is the MATLAB command prompt


if size(XB,1)~=1 | size(YB,1) ~=1
   error ('airfoil specification error-XB & YB must be row vectors -transpose your arrays!')
end

[XMIN,ILE]=min(XB); MP1=size(XB,2); M=MP1-1; ALPHA1=ALPHA*pi/180;

if (YB(1)~=0|YB(MP1)~=0)
   error ('airfoil specification error, Y(0) and Y(end) must be zero')
end

if (XB(1)~=XB(MP1)|YB(ILE)~=0)
    error ('airfoil specification error')
end

XB=XB'; YB=YB'; c=XB(1)-XMIN; XAC=XMIN+c/4; RHS=zeros(MP1,1);
```

```
gamma=zeros(MP1,1); AN=zeros(MP1,MP1); AT=zeros(M,MP1);
THETA=zeros(M,1); X=zeros(M,1); Y=zeros(M,1); S=zeros(M,1);
SX=zeros(M,1); SY=zeros(M,1); SINE=zeros(M,1); COSINE=zeros(M,1);
X=(XB([1:M])+XB([2:MP1]))/2; Y=(YB([1:M])+YB([2:MP1]))/2;
SX=XB([2:MP1])-XB([1:M]); SY=YB([2:MP1])-YB([1:M]);
S=sqrt(SX.^2+SY.^2); THETA=atan2(SY,SX); SINE=sin(THETA);
COSINE=cos(THETA);

for I=1:M
   for J=1:M
      if I==J
         CN1(I,J)=-1.0;
         CN2(I,J)=1.0;
         CT1(I,J)=pi/2;
         CT2(I,J)=pi/2;
     else
         XTEMP=X(I)-XB(J);
         YTEMP=Y(I)-YB(J);
         A=-XTEMP*COSINE(J)-YTEMP*SINE(J);
         B=XTEMP^2+YTEMP^2;
         TTEMP=THETA(I)-THETA(J);
         C=sin(TTEMP);
         D=cos(TTEMP);
         E=XTEMP*SINE(J)-YTEMP*COSINE(J);
         F=log(1+S(J)*(S(J)+2*A)/B);
         G=atan2(E*S(J),B+A*S(J));
         TTEMP=TTEMP-THETA(J);
         P=XTEMP*sin(TTEMP)+YTEMP*cos(TTEMP);
         Q=XTEMP*cos(TTEMP)-YTEMP*sin(TTEMP);
         CN2(I,J)=D+Q*F/(2*S(J))-(A*C+D*E)*G/S(J);
         CN1(I,J)=D*F/2+C*G-CN2(I,J);
         CT2(I,J)=C+P*F/(2*S(J))+(A*D-C*E)*G/S(J);
         CT1(I,J)=C*F/2-D*G-CT2(I,J);
     end
 end
end

%  Compute influence coefficients
AN=[CN1(:,1),CN1(:,[2:M])+CN2(:,[1:M-1]),CN2(:,M);...
      1,zeros(1,M-1),1];
```

```
AT=[CT1(:,1),CT1(:,[2:M])+CT2(:,[1:M-1]),CT2(:,M)];
RHS=[sin(THETA-ALPHA1);0];

% Compute circulation
gamma=AN\RHS;

% Compute velocity
V=cos(THETA-ALPHA1)+AT*gamma;

% Compute pressure coefficient
cp=1-V.^2;

% Compute total circulation
G=S'*(gamma([1:M])+gamma([2:MP1]))/2;

% Compute lift coefficient
cl=4*pi*G/c;

%Compute pitching moment about aerodynamic center
%ISurf=sign([1:M]'-ILE);ISurf(ILE)=1;
cmac=S'*(cp.*((X-XAC).*COSINE+Y.*SINE)/c^2); xp=X; yp=Y;
```