ASEN 3111 - Computational Assignment 03 - Main

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Produces the lift coefficients and pressure distributions about four NACA airfoils and compares them to thin airfoils


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Author: Samuel Razumovskiy
Collaborators: B.Smith
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(Knock knock) House keeping

```
clear,clc,close all
```

Problem 1

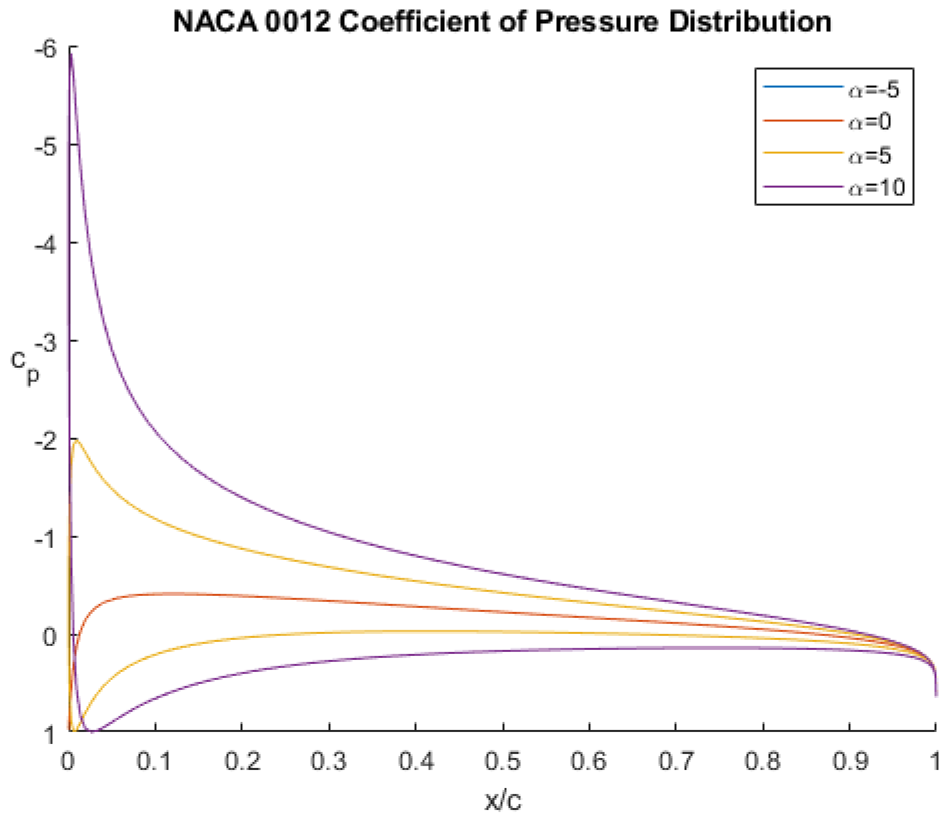
```
% Variations in N
Nset = [24,50,100,150];
% NACA airfoil parameters p, m, and t
m = 0;
p = 0;
t = 12;
c = 1;
alpha  = 0;
v_inf = 50;
tpl = false;
for i = 1:numel(Nset)
    N = Nset(i);
    M = N-2;

    [x,y] = NACA_Airfoils(m,p,t,c,N);
    cl(i) = Vortex_Panel(alpha,v_inf,c,x,y,M,tpl);
end

% N for accurate c_l
N = 500;
M = N-2;
[x,y] = NACA_Airfoils(m,p,t,c,N);
clIdeal = Vortex_Panel(alpha,v_inf,c,x,y,M,tpl);
```

```
error = abs(cl-clIdeal);
fprintf('The error for N panels of 25, 50, 100, and 150\n')
fprintf('with respect to an N panels 500 is\n')
format short g
disp(error')
fprintf('Since using N panels of 150 only comes to an error of about
 1.32e4\n')
fprintf('I will continue to use an N of 150 for the rest of my
 calculations\n\n')
```

```
The error for N panels of 25, 50, 100, and 150
with respect to an N panels 500 is
     0.024858
    0.0050384
     0.000736
   0.00013192

Since using N panels of 150 only comes to an error of about 1.32e4
I will continue to use an N of 150 for the rest of my calculations
```

Problem 2

```
close all

% Variations in alpha
alphas = [-5,0,5,10];
N = 150;
M = N-2;
tpl = true;
figure
hold on
for i = 1:numel(alphas)
    alpha = alphas(i)*pi/180;
    [x,y] = NACA_Airfoils(m,p,t,c,N);
    cl(i) = Vortex_Panel(alpha,v_inf,c,x,y,M,tpl);

    title(sprintf('NACA %1.f%1.f%2.f Coefficient of Pressure
 Distribution',m,p,t))
    xlabel('x/c')
    ylabel('c_p')
    set(get(gca,'ylabel'),'rotation',0)
    leg{i} = sprintf('\\alpha=%1.f',alphas(i));
end
legend(leg)
```

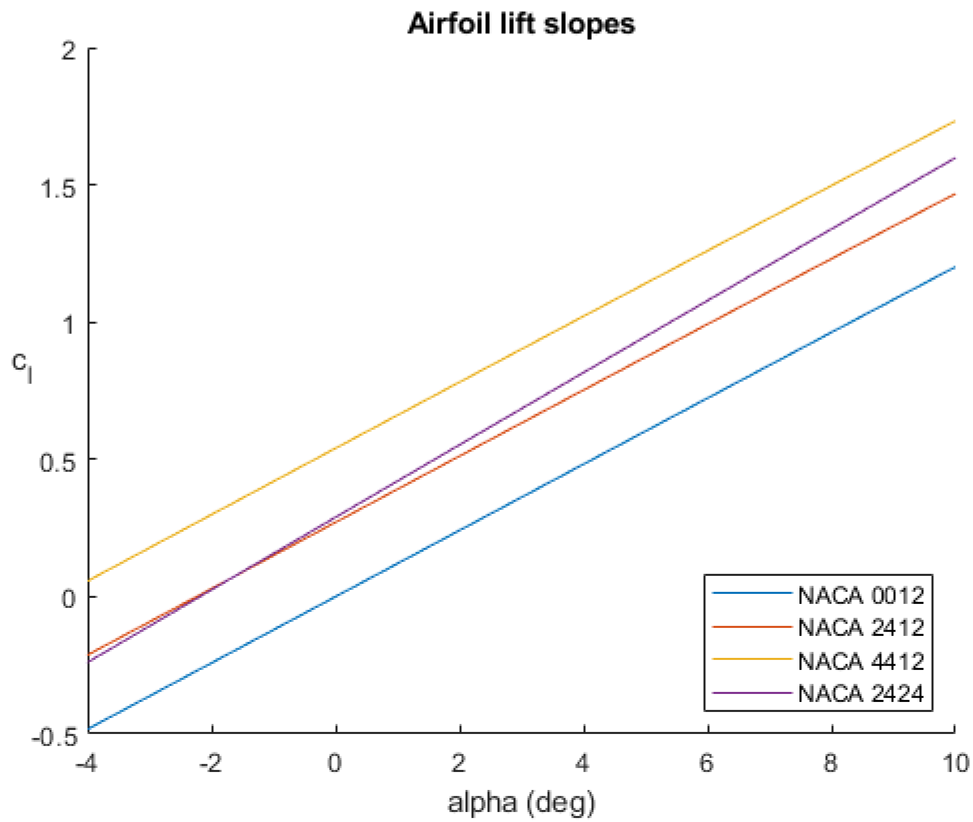


Problem 3

```
close all

% Variations in m, p, and t for each NACA airfoil
ms = [0,2,4,2];
ps = [0,4,4,4];
ts = [12,12,12,24];

% Range of alphas
alphas = linspace(-4,10);
tpl = false;

figure
hold on
for i = 1:numel(ms)
    m = ms(i);
    p = ps(i);
    t = ts(i);
    [x,y,yc] = NACA_Airfoils(m,p,t,c,N);
    for j = 1:numel(alphas)
        alpha = alphas(j)*pi/180;
        [cl(j)] = Vortex_Panel(alpha,v_inf,c,x,y,M,tpl);
    end
    % Calculating the dz_dx
```

```
    dz_dx = (yc(2:end)-yc(1:end-1))./(x(2:N/2)-x(1:N/2-1));
    % Calculating theta
    theta = acos(1-2*x(1:N/2)/c);
    % Finding the midpoints of theta
    thetamid = (theta(2:end)+theta(1:end-1))/2;
    % Calculating the thin airfoil alpha L=0
    thinalpha = 1/pi.*trapz(thetamid,dz_dx.*(cos(thetamid)-1));
    % Finding the slope and y intercept of the thick airfoil data
    fun = polyfit(alphas*pi/180,cl,1);
    % Finding the x intercept of the thick airfoil data
    alphal0 = (-fun(2)/fun(1));
    fprintf('NACA %1.f%1.f%2.f lift slope = %1.2f, alpha L=0 = %1.2f
 deg\n',m,p,t,fun(1),alphal0*180/pi)
    fprintf('Thin airfoil alpha L=0 = %1.2f\n',thinalpha*180/pi)
    plot(alphas,cl)
    leg{i} = sprintf('NACA %1.f%1.f%2.f',m,p,t);
end
title('Airfoil lift slopes')
xlabel('alpha (deg)')
ylabel('c_l')
set(get(gca,'ylabel'),'rotation',0)
legend(leg,'location','southeast')
fprintf('Thin airfoil lift slope is always 6.28\n\n')
fprintf(['The lift slope approximations are similar to the thick
 airfoils '...
    'but obviously smaller, since using a thick airfoil increases lift
\n'])
fprintf(['While the alpha L=0 is almost exactly the same, which makes
 sense'...
    'since the alpha L=0 really only depends on the camber'])
```

```
NACA 0012 lift slope = 6.90, alpha L=0 = 0.00 deg
Thin airfoil alpha L=0 = 0.00
NACA 2412 lift slope = 6.88, alpha L=0 = -2.25 deg
Thin airfoil alpha L=0 = -2.18
NACA 4412 lift slope = 6.86, alpha L=0 = -4.50 deg
Thin airfoil alpha L=0 = -4.35
NACA 2424 lift slope = 7.53, alpha L=0 = -2.20 deg
Thin airfoil alpha L=0 = -2.18
Thin airfoil lift slope is always 6.28

The lift slope approximations are similar to the thick airfoils but
 obviously smaller, since using a thick airfoil increases lift
While the alpha L=0 is almost exactly the same, which makes sensesince
 the alpha L=0 really only depends on the camber
```



Functions Called

The following functions were built and called as part of this assignment.

```
function [x,y,yc] = NACA_Airfoils(m,p,t,c,N)
% NACA_Airfoils produces the x and y coordinates of a desired NACA
 four
% digit airfoil and the mean camber line, provided the four digits
 that
% come from m, p, and t, the chord length, and the number of points N

% Actual values of m, p, and t
m = m/100;
p = p/10;
t = t/100;
% Using the method from Kuthe and Chow to get x values
x = c/2+c/2*cos(linspace(0,pi,ceil(N/2)));

% Y thickness
yt = t.*c/0.2.*(0.2969.*sqrt(x./c)-0.1260.*(x./c)-0.3516.*(x./
c).^2+...
        0.2843.*(x./c).^3-0.1036.*(x./c).^4);

% Y mean camber (This somehow works but I'm not sure if it's more
 efficient
```

```
% than a regular for loop)
if (0 <= x) & (x <= p*c)
    yc = (m/p^2).*x.*(2*p - x./c);
else
    yc = (m/(1-p)^2).*(c - x).*(1 + x./c - 2*p);
end

% Calculating xi
xi =atan(diff(yc));
xi(end+1) = 0;

% Finding X upper and X lower
xU = x-yt.*sin(xi);
xL = x+yt.*sin(xi);

% Finding Y upper and Y lower then concatenating everything
yU = yc+yt.*cos(xi);
yL = yc-yt.*cos(xi);
x = [xL,fliplr(xU(1:end-1))];
y = [yL,fliplr(yU(1:end-1))];
end

function [cl,xc,yc] = Vortex_Panel(alpha,v_inf,c,x,y,M,tpl)
% Vortex_Panel uses vortex method to calculate the coefficient of lift
% around a given airfoil provided the AoA, V infinity, chord length, x
 and
% y locations of the airfoil, total number of points M, and a boolean
 value
% for whether or not to plot

MP1    = M+1;
xc     = zeros(1,M);
yc     = zeros(1,M);
S      = zeros(1,M);
theta  = zeros(1,M);
sine   = zeros(1,M);
cosine = zeros(1,M);
RHS    = zeros(1,M);
Cp     = zeros(1,M);
V      = zeros(1,M);
CN1    = zeros(M,M);
CN2    = zeros(M,M);
CT1    = zeros(M,M);
CT2    = zeros(M,M);
AN     = zeros(M,M);
AT     = zeros(M,M);

for i = 1:M
    ip1       = i+1;
    xc(i)     = 0.5*(x(i)+x(ip1));
    yc(i)     = 0.5*(y(i)+y(ip1));
    S(i)      = sqrt((x(ip1)-x(i))^2 + (y(ip1)-y(i))^2);
    theta(i)  = atan2((y(ip1)-y(i)),(x(ip1)-x(i)));
    sine(i)   = sin(theta(i));
```

```
    cosine(i) = cos(theta(i));
    RHS(i)    = sin(theta(i)-alpha);
end

for i = 1:M
    for j = 1:M
        if i == j
            CN1(i,j) = -1;
            CN2(i,j) = 1;
            CT1(i,j) = 0.5*pi;
            CT2(i,j) = 0.5*pi;

        else
            A = -(xc(i)-x(j))*cosine(j)-(yc(i)-y(j))*sine(j);
            B = (xc(i)-x(j))^2+(yc(i)-y(j))^2;
            C = sin(theta(i)-theta(j));
            D = cos(theta(i)-theta(j));
            E = (xc(i)-x(j))*sine(j)-(yc(i)-y(j))*cosine(j);
            F = log(1+S(j)*(S(j)+2*A)/B);
            G = atan2(E*S(j),B+A*S(j));
            P = (xc(i)-x(j))*sin(theta(i)-2*theta(j))+...
                (yc(i)-y(j))*cos(theta(i)-2*theta(j));
            Q = (xc(i)-x(j))*cos(theta(i)-2*theta(j))-...
                (yc(i)-y(j))*sin(theta(i)-2*theta(j));
            CN2(i,j) = D+.5*Q*F/S(j)-(A*C+D*E)*G/S(j);
            CN1(i,j) = .5*D*F+C*G-CN2(i,j);
            CT2(i,j) = C+.5*P*F/S(j)+(A*D-C*E)*G/S(j);
            CT1(i,j) = .5*C*F-D*G-CT2(i,j);
        end
    end
end

for i=1:M
    AN(i,1) = CN1(i,1);
    AN(i,MP1) = CN2(i,M);
    AT(i,1) = CT1(i,1);
    AT(i,MP1) = CT2(i,M);
    for j=2:M
        AN(i,j) = CN1(i,j)+CN2(i,j-1);
        AT(i,j) = CT1(i,j)+CT2(i,j-1);
    end
end
AN(MP1,1)   = 1;
AN(MP1,MP1) = 1;
for j = 2:M
    AN(MP1,j) = 0;
end
RHS(MP1) = 0;
gamma = AN\RHS';
for i=1:M
    V(i) = cos(theta(i)-alpha);
    for j=1:MP1
        V(i) = V(i)+AT(i,j)*gamma(j);
    end
```

```
    Cp(i) = 1-V(i)^2;
end

% Calclating Gamma
rho = 1.225;
pinf = 101325;
Qinf = .5 * rho * v_inf^2;
S = [S,S(1)];
Gamma = sum(2*pi*v_inf*(gamma'.*S));
cl = 2*Gamma/(v_inf*c);

if tpl == true
    plot(xc/c,Cp)
    set(gca,'YDir','reverse')
end
end
```

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