Airfoil Plotter

- This program will plot the airfoil of a NACA 4-digit series aircraft.

Theory

First of all. the nomenclature of the 4-digit is the following

1. First digit: The **maximum camber** as percentage of chord
2. Second Digit: **Distance of the maximum camber** from the airfoil leading edge in tens of percents of chord
3. Last two digits: The **maximum thickness** of airfoil as percentage of chord

Then, the equations we will use to calculate the airfoil shape are shown below.

The equation for the airfoil is experessed as a pair of parameteric equations for X and Y using the parameter θ for $\theta = [0, 2\pi]$. The equations are

$$X(\theta) = 0.5 + 0.5\frac{|\cos(\theta)|^B}{\cos(\theta)} \quad \cdots (1)$$

$$Y(\theta) = \frac{T}{2}\frac{|\sin(\theta)|^B}{\sin(\theta)}(1 - X^P) + \mathrm{Csin}(X^E\pi) + \mathrm{Rsin}(\mathrm{X}2\pi) \quad \cdots (2)$$

where

1. B: Describes the **Base shape coefficient.** This parameter determines mainly the shape of the leading edge. When B is closer to 2 the base shape of the airfoil becomes more ellliptical, whereas when it is more closer to 1 the shape becomes more rectangular.
2. T: Describes **thickness** as a fraction/percentage of the chord
3. P: Describes the **Taper Exponent**. The more P is closer to 1 the thickness tends to decrease more linearly when approaching 0, whereas the more P is a higher value the thickness decreases more suddenly
4. E: Describes the **Camber Exponent**. This defines the position of the maximum camber point on the chord. Where E = 1 indicates the camber point to be at the middle of the airfoil, that is 50% camber point. Smaller value of E shifts the camber point more toward the leading edge
5. R: Describes the **Reflex Parameter**. When this value is a positive value the trailing edge becomes reflexed, whereas when it is negative one emulates flaps

Next, the Equation to plot the mean chord line will be simply horizontal line connecting $x = [0, 1] \cdots (3)$.

Finally, the equation for the mean camber line will be

$$C_m(\theta) = \mathrm{Csin}(X^E\pi) + \mathrm{Rsin}(\mathrm{X}2\pi) \quad \cdots (4)$$

Now the equations from (1) to (4) is the thoery of this function.

Function

```
function  airfoil_plotter_func(fourDigitCode, B_coeff, Taper_coeff, reflex_para)
```

>>INPUT

1. fourDigitCode: The NACA 4-digit code
2. B_coeff: Base shape coefficient
3. Taper_coeff: Taper exponent
4. reflex_para: Reflex parameter

>>OUTPUT

1. none (only plot)

>>SETUP

```
% Take out the parameters from the 4-digit code
% Hold the code
place_holder = fourDigitCode;
% Maximum camber
C_max = floor(fourDigitCode / 1000);
% Take the modulus of the four digit code for further manipulation
fourDigitCode = mod(fourDigitCode, 1000);
% Position of maximum camber
C_exp = floor(fourDigitCode / 100);
% Repeat modulus operation
fourDigitCode = mod(fourDigitCode, 100);
% Maximum thickness
Th_max = fourDigitCode;

% Default settings for the input parameters
% base shape coefficient
if B_coeff == 0
    B_coeff = 2;
end
% Taper coefficient
if Taper_coeff == 0
    Taper_coeff = 1;
end
% Reflex parameter
if reflex_para == 0
    reflex_para = 0;
end

% Assign simpler varibales to make the subsequent calculations easier
% while fixing the parameters from percent to decimal
C = C_max / 100;
B = B_coeff;
P = Taper_coeff;
E = C_exp / 10;
T = Th_max / 100;
R = reflex_para;
```

```
```

>>CALCULATIONS

```
% The main function will be a for loop to retreive all the points for the plots
% The stepsize/increment for each iteration will of the loop will be
stepsize = 2 * pi / 1000;

% Before the loop we must preallocate the vector including all the x- and y-values
% and point for mean camber line
X_theta = zeros(1000, 1);
Y_theta = zeros(1000, 1);
Cm_theta = zeros(1000,1);

% The loop
% Initialize index counter
n = 1;

for theta = 0:stepsize:2*pi
    % Calculating the x-value
    X = 0.5 + 0.5 * (abs(cos(theta)))^B / cos(theta);
    % Calculating the y-value
    Y = T * (abs(sin(theta)))^B * (1-X^P) / 2 / sin(theta) + C * sin(X^E * pi)...
        + R * sin(X * 2 * pi);
    % Calculating point for mean camber line
    Cm = C * sin(X^E * pi) + R * sin(X * 2 * pi);

    % Apending these values into the vector
    X_theta(n) = X;
    Y_theta(n) = Y;
    Cm_theta(n) = Cm;

    % Increment counter
    n = n + 1;
end
% The chord line will be
X_chord = [0,1];
Y_chord = [0,0];
```

>>PLOTTING

```
figure(1)
plot(X_theta, Y_theta,"-b",'Linewidth', 2.5)
title(['The Airfoil Geometry of NACA' num2str(place_holder)])
xlabel('0 to 1 from Leading Edge to Trailing Edge')
ylabel('Vertical Direction')
grid on
grid minor
ylim([-0.2, 0.2])
box on
hold on
```

```
plot(X_theta, Cm_theta,"-r", 'Linewidth', 1.5)
plot(X_chord, Y_chord, "--g")
hold off
```

Terminate Function

```
end
```