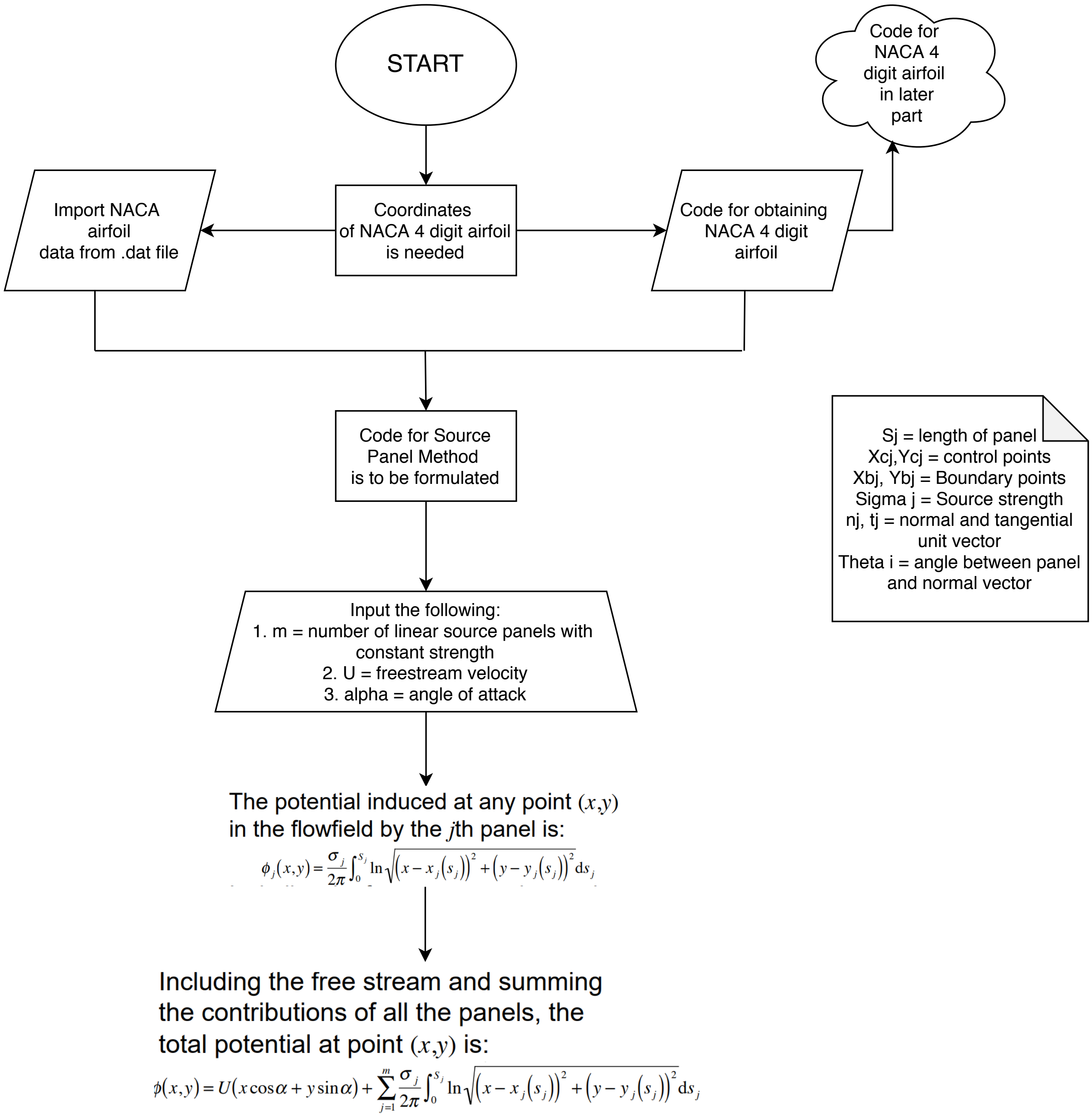
START
Code for NACA 4 digit airfoil in later part
Import NACA airfoil data from .dat file
Coordinates of NACA 4 digit airfoil is needed
Code for obtaining NACA 4 digit airfoil
Code for Source Panel Method is to be formulated
Sj = length of panel
Xcj,Ycj = control points
Xbj, Ybj = Boundary points
Sigma j = Source strength
nj, tj = normal and tangential unit vector
Theta i = angle between panel and normal vector
Input the following:
1. m = number of linear source panels with constant strength
2. U = freestream velocity
3. alpha = angle of attack
The potential induced at any point (x,y) in the flowfield by the jth panel is:
$\phi_j(x,y) = \frac{\sigma_j}{2\pi}\int_0^{S_j} \ln\sqrt{\left(x - x_j(s_j)\right)^2 + \left(y - y_j(s_j)\right)^2}\,\mathrm{d}s_j$
Including the free stream and summing the contributions of all the panels, the total potential at point (x,y) is:
$\phi(x,y) = U(x\cos\alpha + y\sin\alpha) + \sum_{j=1}^{m}\frac{\sigma_j}{2\pi}\int_0^{S_j} \ln\sqrt{\left(x - x_j(s_j)\right)^2 + \left(y - y_j(s_j)\right)^2}\,\mathrm{d}s_j$

Find out the following where Sj is length of jth panel

$$x_j(s_j) = \frac{xb_{j+1} - xb_j}{S_j} s_j + xb_j = \cos\theta_j s_j + xb_j$$

$$y_j(s_j) = \frac{yb_{j+1} - yb_j}{S_i} s_j + yb_j = \sin\theta_j s_j + yb_j$$

The total potential becomes as following

$$\phi(x,y) = U(x\cos\alpha + y\sin\alpha) + \sum_{j=1}^{m} \frac{\sigma_j}{2\pi} \int_0^{S_j} \ln\sqrt{\left(x - xb_j - \cos\theta_j s_j\right)^2 + \left(y - yb_j - \sin\theta_j s_j\right)^2}\, \mathrm{d}s_j$$

Boundary Conditions are applied

$$V_{\infty,n} + V_n = 0$$

Condition is applied on control points and we get following

$$\frac{\partial\phi(xc_i, yc_i)}{\partial n_i} = U\left(\frac{\partial x}{\partial n_i}\cos\alpha + \frac{\partial y}{\partial n_i}\sin\alpha\right) + \sum_{j=1}^{m} \frac{\sigma_j}{2\pi} \int_0^{S_j} \frac{\partial}{\partial n_i}\left(\ln\sqrt{\left(xc_i - xb_j - \cos\theta_j s_j\right)^2 + \left(yc_i - yb_j - \sin\theta_j s_j\right)^2}\right) \mathrm{d}s_j$$

Since we know

$$\frac{\partial x}{\partial n_i} = \sin\theta_i, \frac{\partial y}{\partial n_i} = -\cos\theta_i$$

Thus, we obtain following

$$\sum_{j=1}^{m}\frac{\sigma_j}{2\pi}\int_0^{S_j}\frac{\partial}{\partial n_i}\left(\ln\sqrt{\left(xc_i - xb_j - \cos\theta_j s_j\right)^2 + \left(yc_i - yb_j - \sin\theta_j s_j\right)^2}\right)ds_j = U\sin\left(\theta_i - \alpha\right)$$

We will carry out the integration such that

The boundary condition becomes:

$$\sum_{j=1}^{m}\frac{\sigma_j}{2\pi}\left(-\frac{C_{ij}F_{ij}}{2} + D_{ij}G_{ij}\right) = U\sin\left(\theta_i - \alpha\right)$$

Where:

$$A_{ij} = -\left(xc_i - xb_j\right)\cos\theta_j - \left(yc_i - yb_j\right)\sin\theta_j$$

$$B_{ij} = \left(xc_i - xb_j\right)^2 + \left(yc_i - yb_j\right)^2$$

$$C_{ij} = \sin\left(\theta_i - \theta_j\right),\ \ D_{ij} = \cos\left(\theta_i - \theta_j\right),\ F = \ln\left(1 + \frac{S_j^2 + 2A_{ij}S_j}{B_{ij}}\right)$$

$$E_{ij} = \left(xc_i - xb_j\right)\sin\theta_j - \left(yc_i - yb_j\right)\cos\theta_j,\ \ G_{ij} = \tan^{-1}\left(\frac{E_{ij}S_j}{A_{ij}S_j + B_{ij}}\right)$$

$$-\frac{C_{ii}F_{ii}}{2} + D_{ii}G_{ii} = -\pi$$

The tangential velocity is given as

$$v_{t_i} = \frac{\partial\phi\left(xc_i, yc_i\right)}{\partial t_i}$$

Thus, coefficient of pressure is obtained as follows

$$c_{p_i} = 1 - \left(\frac{v_{t_i}}{U}\right)^2$$

Code for Airfoil NACA 4 digit

1st digit = maximum camber as a percentage of chord
2nd digit = distance of maximum camber from the airfoil leading edge in tenths of the chord
3rd & 4th digits: maximum thickness of airfoil as percent of chord

Symmetric airfoil

$$y_t = 5t\left[0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right]$$

where:

x is the position along the chord from 0 to 1.00 (0 to 100%),

y_t is the half thickness at a given value of *x* (centerline to surface),

t is the maximum thickness as a fraction of the chord

$$x_U = x_L = x$$
$$y_U = +y_t$$
$$y_L = -y_t$$

Cambered airfoil

$$y_c = \begin{cases} \dfrac{m}{p^2}\left(2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & 0 \le x \le pc, \\ \dfrac{m}{(1-p)^2}\left((1-2p) + 2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & pc \le x \le c, \end{cases}$$

m is the maximum camber

p is the location of maximum camber

$$x_U = x - y_t\,\sin\theta, \qquad y_U = y_c + y_t\,\cos\theta$$
$$x_L = x + y_t\,\sin\theta, \qquad y_L = y_c - y_t\,\cos\theta$$

(Xu,Yu) and (XL,YL) are coordinates of upper and lower airfoil surface