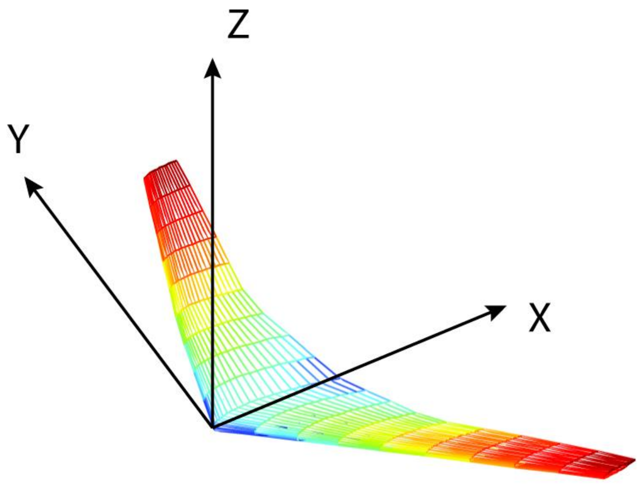
Guidelines for using Q3D AeroSolver

1. **Wing planform geometry definition:**
    a. The coordinates of the apex points (x,y,z) should be defined in meter.
    b. The chord should be defined in meter.
    c. The twist angle should be defined in degree (positive = nose up).
    d. The axis (for defining x, y, and z) is illustrated in the following figure:



    e. Only the left wing (from front view) should be defined. The other wing will be automatically created by mirroring the defined wing.

2. **Airfoil geometry definition:**
    a. All the airfoils should be defined with the same number of CST coefficients.
    b. The number of CST coefficient for the upper and lower surfaces must be the same.

3. **Viscous vs inviscid analysis**
    a. The wing analysis can be switched between viscous and inviscid analysis using AC.Visc variable. If the value of this variable is chosen to be 1 the Q3D solver performs a viscous analysis and reports the total wing viscous drag.
    b. If the value of AC.Visc is chosen to be 0 the Q3D solver only performs an inviscid analysis and the outputs contains lift, pitching

moment and only induced drag. This analysis is useful when only the lift and pitching moment distribution are needed.

4. **Flow condition:**
    a. Even if some of the input parameters are dependent variables (for instance, V can be calculated when M and H are given), ALL of them must be provided, hence it is user responsibility to provide a consistent set.
    b. The values of speed, altitude, density and viscosity must be expressed in SI units (m, kg, s).

5. **Tool installation and system requirement:**
    a. Q3D AeroSolver should be used in Matlab 2009 or higher.
    b. The tool is only available for Windows (that is because, the commercial software VGK used by Q3D AeroSolver is only available for Windows).
    c. When installing the tool on your PC, do NOT change file and folder names and structure.

6. **The Q3D AeroSolver is developed only for wing analysis. The effect of fuselage and tail cannot be modelled in this tool.**

7. **The outputs (Res):**
    a. *Res.Alpha* is the 3D wing angle of attack. The angle of attack of each wing spanwise section is the wing angle of attack plus the wing incidence (if defined) plus the local twist angle.
    b. *Res.CLwing* is the 3D wing total lift coefficient.
    c. *Res.CDwing* is the 3D wing total drag coefficient (including profile, induced and wave drag if AC.Visc =1 but **only** induced drag if AC.Visc =0).

    d. *Res.Wing* includes the results of VLM analysis:
        i. *Res.Wing.Yst* is the spanwise location of each strips (where the vortexes are placed).
        ii. *Res.Wing.chord* is the wing chord at each strip.
        iii. *Res.Wing.cl* is the local lift coefficient at each strip.

iv. *Res.Wing.ccl* is the local lift coefficient multiplied by the local chord at each strip.
v. *Res.Wing.cdi* is the local induced drag coefficient at each strip.
vi. *Res.Wing.cm_c4* is the local pitching moment coefficient about the quarter chord line at each strip. This pitching moment includes the moment due to the wing sweep and the moment due to the airfoils camber, but the effect of airfoil thickness is not taken into account. It also does not contain the effect of shock waves if they exist.

e. *Res.Section* includes the results of 2D sections analysis (only for viscous analysis, for inviscid analysis section properties are not reported):
    i. *Res.Section.Y* is the spanwise locations of the analysed 2D sections.
    ii. *Res.Section.Cl* is the local lift coefficient of the analysed 2D sections.
    iii. *Res.Section.Cd* is the drag coefficient (profile + wave drag) of the analysed 2D sections.
    iv. *Res.Section.Cm* is the pitching moment coefficient of the analysed 2D sections around the airfoil quarter chord. This pitching includes the effect of airfoil shape (both camber and thickness) and also includes the effects of compressibility (shock waves if they exist).

Example of Matlab input file for Q3D

```
%% Aerodynamic solver setting

% Wing planform geometry
%                x     y      z    chord(m)    twist angle (deg)
AC.Wing.Geom = [0      0      0     3.5          0;
                0.9   14.5    0     1.4          0];

% Wing incidence angle (degree)
AC.Wing.inc  = 0;


% Airfoil coefficients input matrix
%                    | ->     upper curve coeff.    <-| | ->      lower curve coeff.       <-|
AC.Wing.Airfoils =[0.2171 0.3450 0.2975 0.2685 0.2893 -0.1299 -0.2388 -0.1635 -0.0476 0.0797;
                   0.2171 0.3450 0.2975 0.2685 0.2893 -0.1299 -0.2388 -0.1635 -0.0476 0.0797];

AC.Wing.eta = [0;1];  % Spanwise location of the airfoil sections

% Viscous vs inviscid
AC.Visc  = 1;              % 0 for inviscid and 1 for viscous analysis

% Flight Condition
AC.Aero.V     = 68;            % flight speed (m/s)
AC.Aero.rho   = 1.225;         % air density  (kg/m3)
AC.Aero.alt   = 0;             % flight altitude (m)
AC.Aero.Re    = 1.14e7;        % reynolds number (bqased on mean
aerodynamic chord)
AC.Aero.M     = 0.2;           % flight Mach number
% AC.Aero.CL    = 0.4;          % lift coefficient - comment this line to
run the code for given alpha%
AC.Aero.Alpha = 2;             % angle of attack -  comment this line to
run the code for given cl


Res = Q3D_solver(AC);
```

Example of outputs of Q3D

```
Res =

     Alpha: 2                        % Wing total angle of attack
    CLwing: 0.5967                   % wing total CL
    CDwing: 0.0143                   % wing Total CD
    CMwing: -0.3577                  % Wing total CM
      Wing: [1x1 struct]             % Results of VLM (see below)
   Section: [1x1 struct]             % Results of airfoil analysis (see below)


>> Res.Wing

ans =

     Yst: [14x1 double]
   chord: [14x1 double]
      cl: [14x1 double]
     ccl: [14x1 double]
     cdi: [14x1 double]
   cm_c4: [14x1 double]


>> Res.Section

ans =

    Y: [0 2.0714 4.1429 6.2143 8.2857 10.3571 12.4286 14.5000]
   Cl: [0.5592 0.5861 0.6053 0.6183 0.6251 0.6214 0.5851 0.4255]
   Cd: [0.0047 0.0048 0.0048 0.0049 0.0050 0.0051 0.0051 0.0051]
   Cm: [-0.1119 -0.1120 -0.1120 -0.1119 -0.1117 -0.1113 -0.1107 -0.1089]
```