Aircraft Data for the Real-Time Flight Simulation

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Author: Marek M. Cel

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Notation

$a=\frac{dC_Z}{d\alpha}$, $a=\frac{dC_L}{d\alpha}$	–	[1/rad] lift curve slope
$A=\frac{b^2}{S}$	–	[-] wing aspect ratio
b	–	[m] wing span
c_{root}	–	[m] chord at wing root
c_{tip}	–	[m] chord at wing tip
$\hat{c}$, c_{MAC}	–	[m] mean aerodynamic chord
C_l	–	[-] rolling moment coefficient
C_m	–	[-] pitching moment coefficient
C_n	–	[-] yawing moment coefficient
C_X, C_D	–	[-] drag coefficient
C_Y	–	[-] side force coefficient
C_Z, C_L	–	[-] lift coefficient
C_μ	–	[-] k-ε turbulence model constant
d	–	[m] fuselage diameter
D	–	[N] drag
e	–	[-] Oswald efficiency factor
i	–	[rad] incidence angle
I	–	[-] turbulence intensity
k	–	[m^2/s^2] turbulence kinetic energy
L	–	[N] lift
L	–	[m] reference length scale
p	–	[Pa] pressure
Re	–	[-] Reynolds number
S	–	[m^2] wing area
V	–	[m/s] velocity
α	–	[rad] angle of attack
β	–	[rad] angle of sideslip
$\lambda=\frac{c_{tip}}{c_{root}}$	–	[-] wing taper ratio
Λ_{LE}	–	[rad] leading edge sweep angle
$\Lambda_{t/c}$	–	[rad] sweep angle at maximum thickness
ω	–	[1/s] specific turbulence dissipation rate
$\frac{\partial\epsilon}{\partial\alpha}$	–	[-] horizontal stabilizer downwash angle derivative with respect to the aircraft angle of attack

1. Geometric Parameters

1.1. Wing Basic Geometric Parameters

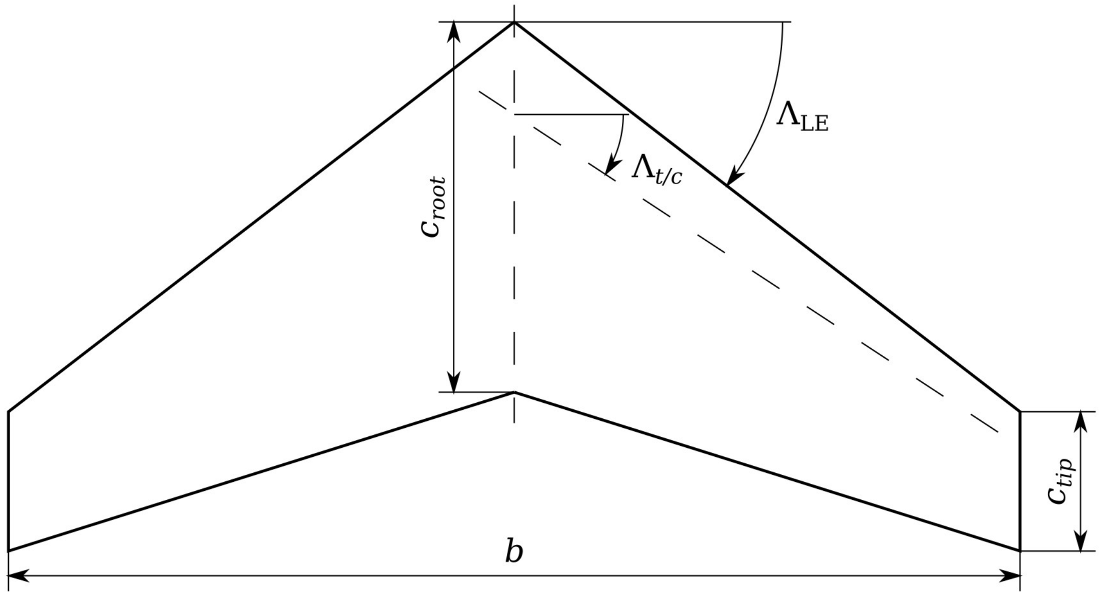

Figure 1-1: Wing basic geometric parameters

Aspect ratio is given by the following formula. [1]

$$A=\frac{b^2}{S} \tag{1.1}$$

Taper ratio is given by the following formula. [1]

$$\lambda=\frac{c_{tip}}{c_{root}} \tag{1.2}$$

1.2. Mean Aerodynamic Chord

For tapper wing mean aerodynamic chord can be calculated using following formula. [2]

$$\hat{c}=\frac{2}{3}c_{root}\frac{1+\lambda+\lambda^2}{1+\lambda} \tag{1.3}$$

For more complex shapes mean aerodynamic chord is given as follows. []

$$\hat{c}=\left(\int_{-\frac{b}{2}}^{\frac{b}{2}}\left(c\left(y\right)\right)^2 dy\right)\div\left(\int_{-\frac{b}{2}}^{\frac{b}{2}} c\left(y\right)dy\right) \tag{1.4}$$

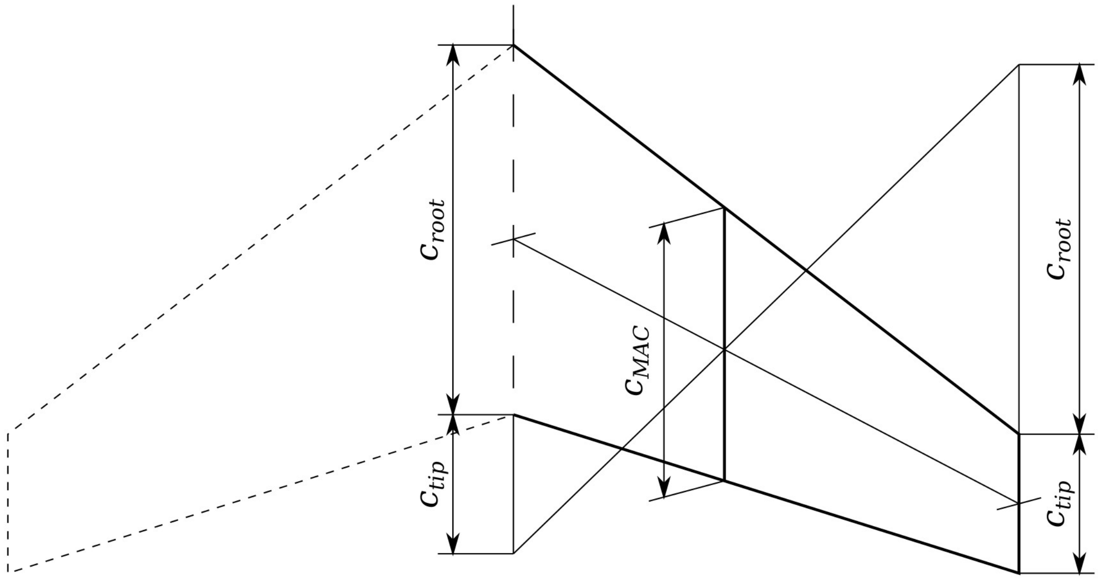

Figure 1-2: Mean aerodynamic chord

1.3. Wing Aerodynamic Center

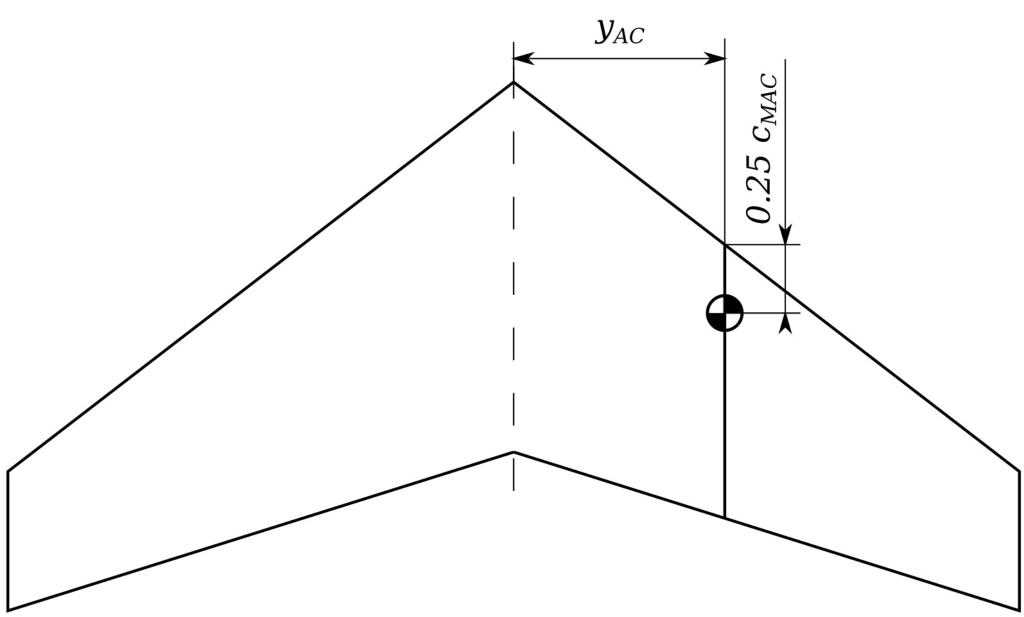

Figure 1-3: Wing aerodynamic center

Position of wing aerodynamic center $\vec{r}_{AC}$ is at 25% of the mean aerodynamic chord and its lateral coordinate is given by the following formula. [1], [2], [3]

$$y_{AC} = \frac{b(1+2\lambda)}{6(1+\lambda)} \tag{1.5}$$

2. Aerodynamic Characteristics

2.1. Aerodynamic Characteristics Approximation

Following approximation is used to get lift coefficient for the full range of angle of attack. [4]

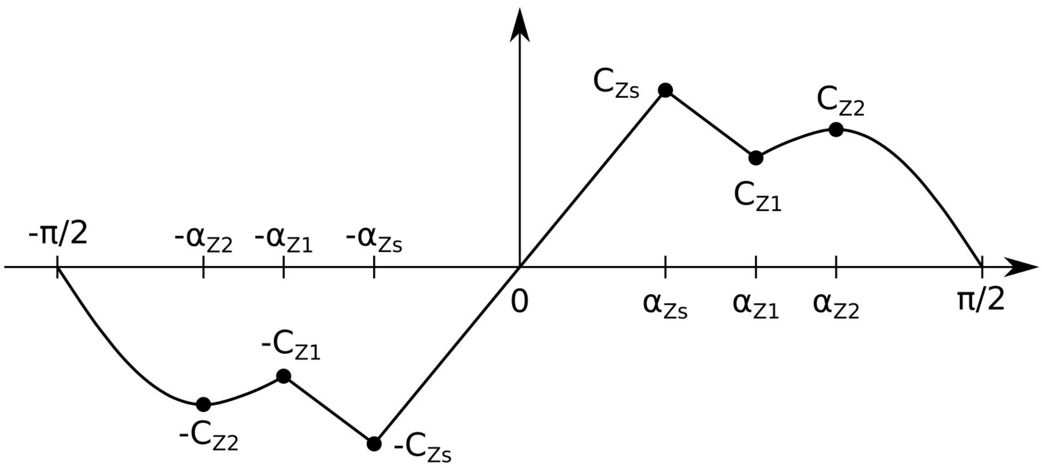

Figure 2-1: Lift coefficient approximation

Lift coefficient is given by the following expressions.

$$C_Z=\frac{-(\alpha+\alpha_{Z2})\left(\alpha+\frac{\pi}{2}\right)}{(\alpha_{Z1}-\alpha_{Z2})\left(\alpha_{Z1}-\frac{\pi}{2}\right)}C_{Z1}-\frac{(\alpha+\alpha_{Z1})\left(\alpha+\frac{\pi}{2}\right)}{(\alpha_{Z2}-\alpha_{Z1})\left(\alpha_{Z2}-\frac{\pi}{2}\right)}C_{Z2}\text{, for }-\frac{\pi}{2}\leqslant\alpha\leqslant-\alpha_{Z1} \quad (2.1)$$

$$C_Z=\frac{C_{Z1}-C_{Zs}}{\alpha_{Z1}-\alpha_{Zs}}(\alpha+\alpha_{Zs})-C_{Zs}\text{, for }-\alpha_{Z1}<\alpha\leqslant-\alpha_{Zs} \quad (2.2)$$

$$C_Z=\frac{C_{Zs}}{\alpha_{Zs}}\alpha\text{, for }-\alpha_{Zs}<\alpha<\alpha_{Zs} \quad (2.3)$$

$$C_Z=\frac{C_{Z1}-C_{Zs}}{\alpha_{Z1}-\alpha_{Zs}}(\alpha-\alpha_{Zs})+C_{Zs}\text{, for }\alpha_{Zs}\leqslant\alpha<\alpha_{Z1} \quad (2.4)$$

$$C_Z=\frac{(\alpha-\alpha_{Z2})\left(\alpha-\frac{\pi}{2}\right)}{(\alpha_{Z1}-\alpha_{Z2})\left(\alpha_{Z1}-\frac{\pi}{2}\right)}C_{Z1}+\frac{(\alpha-\alpha_{Z1})\left(\alpha-\frac{\pi}{2}\right)}{(\alpha_{Z2}-\alpha_{Z1})\left(\alpha_{Z2}-\frac{\pi}{2}\right)}C_{Z2}\text{, for }\alpha_{Z1}\leqslant\alpha\leqslant\frac{\pi}{2} \quad (2.5)$$

Following approximation is used to get drag coefficient for the full range of angle of attack. [4] Drag coefficient is assumed to be symmetric.

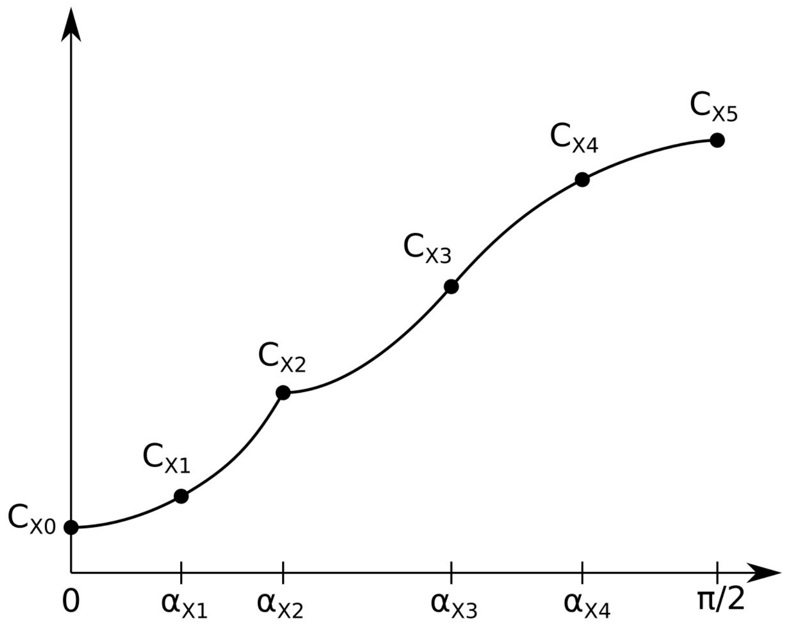

Figure 2-2: Drag coefficient approximation

Drag coefficient is given by the following expressions.

$$C_X = \frac{\left(\alpha^2-\alpha_{X2}^2\right)\left(\alpha^2-\alpha_{X1}^2\right)}{\alpha_{X1}^2\alpha_{X2}^2} C_{X0} + \frac{\left(\alpha^2-\alpha_{X2}^2\right)\alpha^2}{\left(\alpha_{X1}^2-\alpha_{X2}^2\right)\alpha_{X1}^2} C_{X1} + \frac{\left(\alpha^2-\alpha_{X1}^2\right)\alpha^2}{\left(\alpha_{X2}^2-\alpha_{X1}^2\right)\alpha_{X2}^2} C_{X2}, \quad \text{for } -\alpha_{X2} \leqslant \alpha \leqslant \alpha_{X2} \tag{2.6}$$

$$\begin{aligned} C_X = {} & \frac{(\alpha-\alpha_{X3})(\alpha-\alpha_{X4})\left(\alpha-\frac{\pi}{2}\right)}{(\alpha_{X2}-\alpha_{X3})(\alpha_{X2}-\alpha_{X4})\left(\alpha_{X2}-\frac{\pi}{2}\right)} C_{X2} + \frac{(\alpha-\alpha_{X2})(\alpha-\alpha_{X4})\left(\alpha-\frac{\pi}{2}\right)}{(\alpha_{X3}-\alpha_{X2})(\alpha_{X3}-\alpha_{X4})\left(\alpha_{X3}-\frac{\pi}{2}\right)} C_{X3} \\ & + \frac{(\alpha-\alpha_{X2})(\alpha-\alpha_{X3})\left(\alpha-\frac{\pi}{2}\right)}{(\alpha_{X4}-\alpha_{X2})(\alpha_{X4}-\alpha_{X3})\left(\alpha_{X4}-\frac{\pi}{2}\right)} + C_{X4} \frac{(\alpha-\alpha_{X2})(\alpha-\alpha_{X3})(\alpha-\alpha_{X4})}{\left(\frac{\pi}{2}-\alpha_{X2}\right)\left(\frac{\pi}{2}-\alpha_{X3}\right)\left(\frac{\pi}{2}-\alpha_{X4}\right)} C_{X5}, \\ & \text{for } -\alpha_{X2} \leqslant \alpha \leqslant \alpha_{X2} \end{aligned} \tag{2.7}$$

Data available in [5] and [6] can be used to approximate aerodynamic characteristics outside linear range of the lift.

2.2. Finite Wing Aerodynamic Characteristics

Aerodynamic characteristics of the finite wing can be estimated within linear range of the lift.

Lift curve slope is given as follows. [7]

$$\frac{dC_L}{d\alpha}=\frac{2\pi A}{2+\sqrt{4+A^2\left(1+\tan^2\left(\Lambda_{t/c}\right)\right)}} \tag{2.8}$$

The finite wing lift coefficient within linear range is given by the following formula. [7]

$$C_L=C_{L\,\alpha=0}+\frac{d\,C_L}{d\,\alpha}\alpha \tag{2.9}$$

The finite wing maximum lift coefficient is given by. [1]

$$C_{Lmax}=0.9\,C_{Lmax\infty}\cos\Lambda_{t/c} \tag{2.10}$$

The finite wing drag coefficient is given as follows. [7]

$$C_D=C_{D0}+\frac{C_L^2}{\pi A e} \tag{2.11}$$

2.3. Horizontal Tail Incidence

Equilibrium of moments acting on an aircraft is given by the following equation.

$$r_{CG}\,m\,g+\frac{1}{2}\rho V^2 S\hat{c}\,C_m=l_h\frac{1}{2}\rho V^2 S_h C_{Z,h} \tag{2.12}$$

Horizontal stabilizer lift coefficient is given as follows.

$$C_{Z,h}=\left(\alpha+i_h-\alpha\frac{\partial\epsilon}{\partial\alpha}\right)\frac{dC_{Z,h}}{d\,\alpha} \tag{2.13}$$

where downwash derivative is given as

$$\frac{\partial\epsilon}{\partial\alpha}=\frac{2a}{\pi\Lambda} \tag{2.14}$$

Substituting equation (2.13) into (2.12) gives

$$r_{CG}\,m\,g+\frac{1}{2}\rho V^2 S\hat{c}\,C_m=l_h\frac{1}{2}\rho V^2 S_h\left(\alpha+i_h-\alpha\frac{\partial\epsilon}{\partial\alpha}\right)\frac{dC_{Z,h}}{d\,\alpha} \tag{2.15}$$

Solving this equation for horizontal tail incidence angle gives

$$i_h=\frac{2\,r_{CG}\,m\,g+\rho V^2 S\hat{c}\,C_m}{l_h\rho V^2 S_h\frac{dC_{Z,h}}{d\,\alpha}}-\alpha\left(1-\frac{\partial\epsilon}{\partial\alpha}\right) \tag{2.16}$$

Equilibrium of forces acting on an aircraft in level flight is given by the following equation.

$$mg=\frac{1}{2}\rho V^2 S C_Z \tag{2.17}$$

Aircraft lift coefficient is given as follows.

$$C_Z=C_{Z0}+\alpha\frac{dC_Z}{d\alpha} \tag{2.18}$$

Substituting equation (2.18) into (2.17) gives

$$mg=\frac{1}{2}\rho V^2 S\left(C_{Z0}+\alpha\frac{dC_Z}{d\alpha}\right) \tag{2.19}$$

Solving this equation for angle of attack gives

$$\alpha=\frac{2mg-\rho V^2 S C_{Z0}}{\rho V^2 S\frac{dC_Z}{d\alpha}} \tag{2.20}$$

Substituting equation (2.20) into (2.16) gives

$$i_h=\frac{2r_{CG}mg+\rho V^2 S\hat{c}C_m}{l_h\rho V^2 S_h\frac{dC_{Z,h}}{d\alpha}}-\frac{2mg-\rho V^2 S C_{Z0}}{\rho V^2 S\frac{dC_Z}{d\alpha}}\left(1-\frac{\partial\epsilon}{\partial\alpha}\right) \tag{2.21}$$

2.4. Critical Angle of Attack

Equilibrium of forces acting on an aircraft in level flight is given by the following equation.

$$mg=\frac{1}{2}\rho V^2\left(S C_Z+S_h C_{Z,h}\right) \tag{2.22}$$

As conventional configuration airplanes have horizontal stabilizer negative incidence angle, it is assumed that horizontal stabilizer is within its lift linear range when maximum lift coefficient is reached.

$$mg=\frac{1}{2}\rho V^2\left[S C_Z+S_h\left(\alpha_{cr}+i_h-\alpha_{cr}\frac{\partial\epsilon}{\partial\alpha}\right)\frac{dC_{Z,h}}{d\alpha}\right] \tag{2.23}$$

Solving this equation for the maximum lift coefficient gives

$$C_{Z,max}=\frac{mg-\frac{1}{2}\rho V_{stall}^2 S_h\left(\alpha_{cr}+i_h-\alpha_{cr}\frac{\partial\epsilon}{\partial\alpha}\right)\frac{dC_{Z,h}}{d\alpha}}{\frac{1}{2}\rho V_{stall}^2 S} \tag{2.24}$$

Assuming that maximum lift coefficient is within linear range, then critical angle of attack is given as follows.

$$C_{Z,max} = C_{Z0} + \alpha_{cr} \frac{dC_Z}{d\alpha} \tag{2.25}$$

Substituting equation (2.25) into (2.24) gives

$$C_{Z0} + \alpha_{cr} \frac{dC_Z}{d\alpha} = \frac{mg - \frac{1}{2}\rho V_{stall}^2 S_h \left(\alpha_{cr} + i_h - \alpha_{cr} \frac{\partial \epsilon}{\partial \alpha} \right) \frac{dC_{Z,h}}{d\alpha}}{\frac{1}{2}\rho V_{stall}^2 S} \tag{2.26}$$

Solving this equation for critical angle of attack gives

$$\alpha_{cr} = \frac{2mg - \rho V_{stall}^2 \left(S_h i_h \frac{dC_{Z,h}}{d\alpha} + S C_{Z0} \right)}{\rho V_{stall}^2 \left[S \frac{dC_Z}{d\alpha} + S_h \left(1 - \frac{\partial \epsilon}{\partial \alpha} \right) \frac{dC_{Z,h}}{d\alpha} \right]} \tag{2.27}$$

3. Mass and Inertia Data

3.1. Structure Groups Breakdown

Inertia tensor and center of mass coordinates can be estimated by breaking down empty aircraft into structure groups and then estimating their weights, inertia moments and coordinates, e.g. using aircraft drawing. [3]

3.2. OpenVSP

Mass Properties tool of the OpenVSP can be used to compute aircraft center of mass position and inertia tensor.

4. Computational Fluid Dynamics

4.1. Tools

4.1.1. XFOIL

XFOIL is a program for the analysis of subsonic isolated airfoils developed at the Massachusetts Institute of Technology.

4.1.2. VSPAERO

VSPAERO is a combined vortex lattice method (VLM) and panel method solver integrated with OpenVSP, a parametric aircraft geometry tool developed at NASA Ames Research Center. VSPAERO can be used to compute aircraft aerodynamic characteristics within linear range of the lift. [8] OpenVSP allows users to fast create an aircraft 3D model by defining geometric parameters.

4.1.3. OpenFOAM

OpenFOAM is an open source software for computational fluid dynamics (CFD) originally created at Imperial College London. OpenFOAM contains various solvers intended to simulate different physical phenomena.

4.2. Workflow

4.2.1. XFOIL

Follow this steps to obtain airfoil characteristics. Notice that XFOIL commands are case sensitive. [9]

1. Open terminal and execute `xfoil` program.
   XFOIL command promt should start.
2. Read buffer airfoil from coordinate file with `LOAD` command e.g `LOAD 2412.dat`,
   or set NACA 4 or 5 digit airfoil with `NACA` command e.g. `NACA 2412`.
   Coordinate file is a simple text file which contains list of the 2D airfoil coordinate points starting at the trailing edge, progressing to the leading edge along the upper surface, and returning to the trailing edge along the lower surface.
3. Set number of panel nodes to at least 160 with `PPAR` command if necessary.

4. Enter operation mode with `OPER` command. `OPERi` indicates inviscid mode.
5. Enter viscous mode with `Visc` command.
   Enter Reynolds number for typical flight conditions. `OPERv` should be displayed.
6. Set Mach number for typical flight conditions with `Mach` command.
7. Specify output files with `Pacc` command.
8. Specify the angle of attack range and run computations with `Aseq` command.

4.2.2. VSPAERO

An appropriately define aircraft 3D model is needed to analyze. [10] Based on this model degenerated geometry file is generated for the vortex lattice method and surface triangulation file for the panel method.

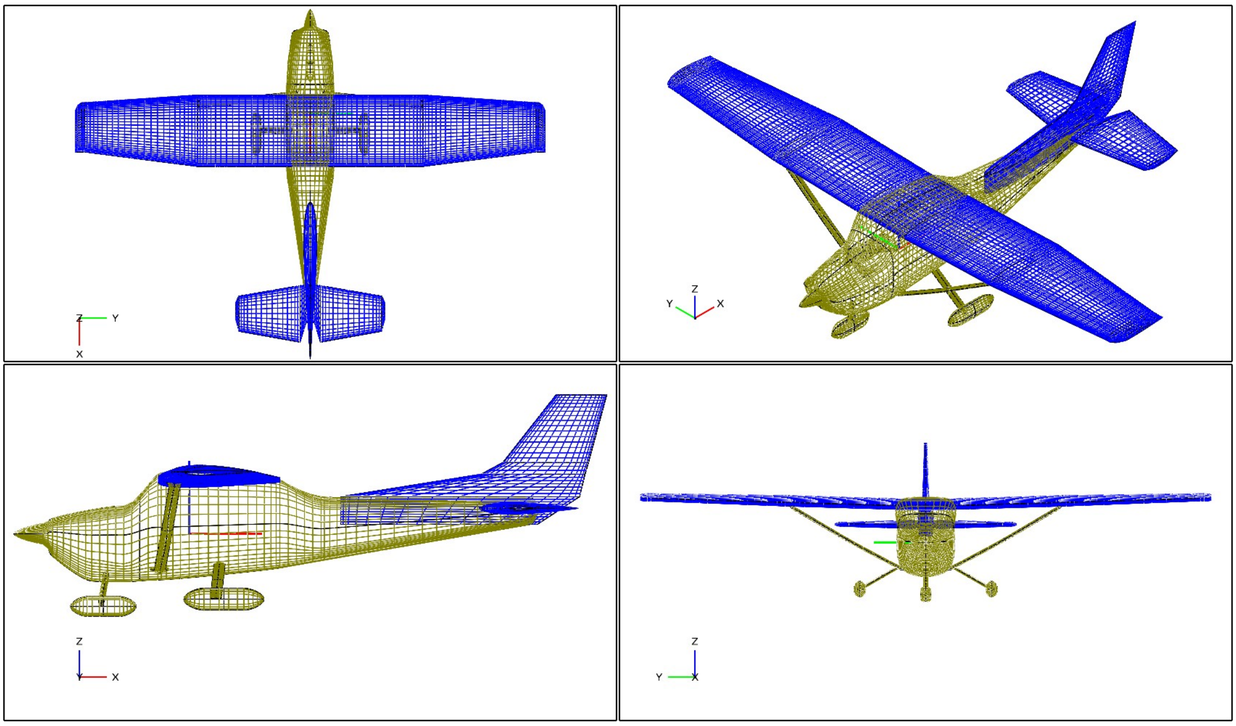

Figure 4-1: OpenVSP aircraft model

VSPAERO can be run from within Analysis menu of the OpenVSP. Some additional setup should be done to start computations. Vortex lattice method or panel method solver should be chosen. Reference area, lengths and moment reference position should be specified to get correct results. Angle of attack range should be specified for the linear range of the lift. Critical angle of attack can be calculated using formula (2.27). Mach number should be set to typical flight conditions.

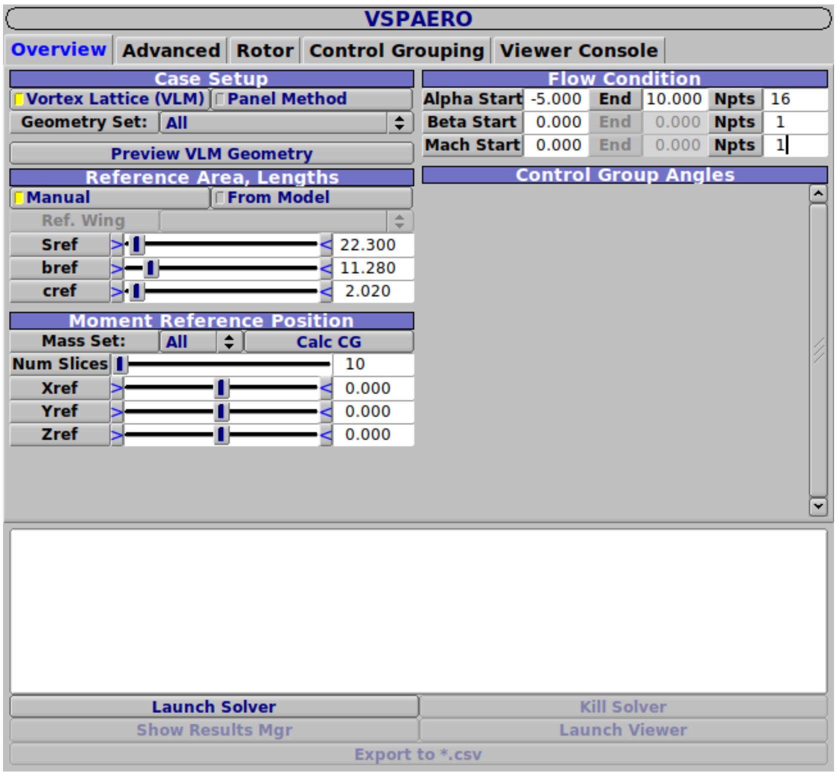

Figure 4-2: VSPAERO computations setup window

VSPAERO Viewer and Results Manager can be used to visualize results. Computed aerodynamic characteristics are saved as plain text files, which are both easy to understand by a human and easy to read by a computer program.

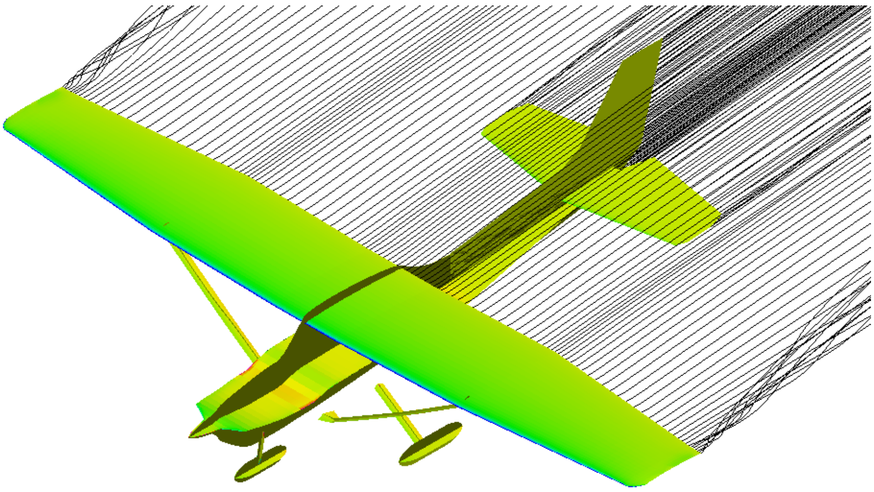

Figure 4-3: Wakes and pressure coefficient change distribution

4.2.3. OpenFOAM

Solver

OpenFOAM `simpleFoam` is a steady-state solver for incompressible, turbulent flow using SIMPLE (Semi-Implicit Method for Pressure Linked Equations) algorithm, which can be used to compute aircraft aerodynamic characteristics for the full range of angle of attack. [11], [12], [13]

SST k-ω Reynolds-Averaged Navier-Stokes (RANS) turbulence model is used.

Mesh – Control Volume

OpenFOAM `blockMesh` utility is used to create control volume mesh defined in `system/blockMeshDict` dictionary file as a mesh composed of hexahedral blocks.

The simplest case is when the control volume is defined by exactly one rectangular prisms.

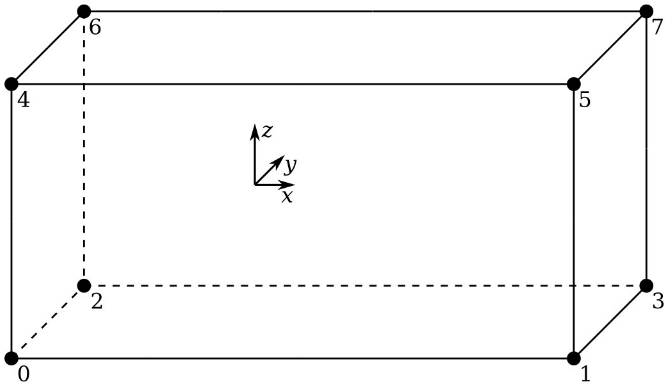

Figure 4-4: Basic control volume scheme

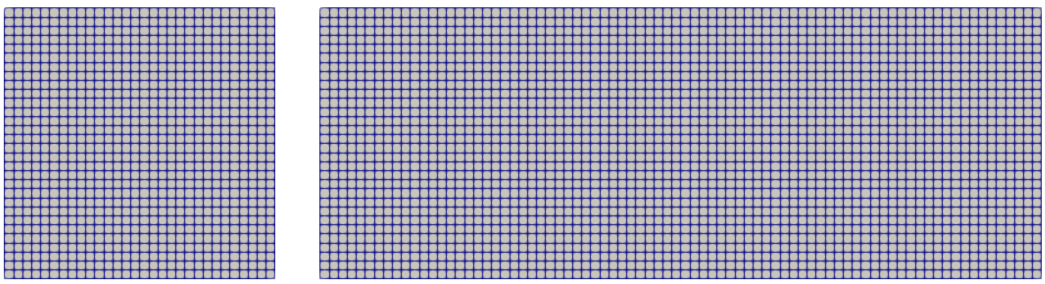

Figure 4-5: Basic control volume mesh

Dictionary file is listed below.

```
/*--------------------------------*- C++ -*----------------------------------*\
| =========                 |                                                 |
| \\      /  F ield         | OpenFOAM: The Open Source CFD Toolbox           |
|  \\    /   O peration     | Version:  6                                     |
|   \\  /    A nd           | Web:      www.OpenFOAM.org                      |
|    \\/     M anipulation  |                                                 |
\*---------------------------------------------------------------------------*/
FoamFile
{
    version     2.0;
    format      ascii;
    class       dictionary;
    object      blockMeshDict;
}
// * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * //

cv_x_min -20.0;
cv_x_max  60.0;
cv_y_min -15.0;
cv_y_max  15.0;
cv_z_min -15.0;
cv_z_max  15.0;

convertToMeters 1.0;

vertices
(
    ( $cv_x_min $cv_y_min $cv_z_min )
    ( $cv_x_max $cv_y_min $cv_z_min )
    ( $cv_x_min $cv_y_max $cv_z_min )
    ( $cv_x_max $cv_y_max $cv_z_min )
    ( $cv_x_min $cv_y_min $cv_z_max )
    ( $cv_x_max $cv_y_min $cv_z_max )
    ( $cv_x_min $cv_y_max $cv_z_max )
    ( $cv_x_max $cv_y_max $cv_z_max )
);

blocks
(
    hex ( 0 1 3 2 4 5 7 6 ) ( 80 30 30 ) simpleGrading ( 1 1 1 )
);

edges
(
);

boundary
(
    inlet
    {
        type patch;
        faces
        (
            ( 0 4 6 2 )
        );
    }
```

```
    outlet
    {
        type patch;
        faces
        (
            ( 1 3 7 5 )
        );
    }

    walls
    {
        type patch;
        faces
        (
            ( 0 1 5 4 ) // left
            ( 2 6 7 3 ) // right
            ( 0 2 3 1 ) // bottom
            ( 4 5 7 6 ) // top
        );
    }
);

mergePatchPairs
(
);
```

Mesh – Simple Grading

Grading is used to increase mesh resolution in regions of interest. Due to create graded control volume mesh it has to be divided into subdomains. Scheme of control volume mesh one dimension grading is presented below.

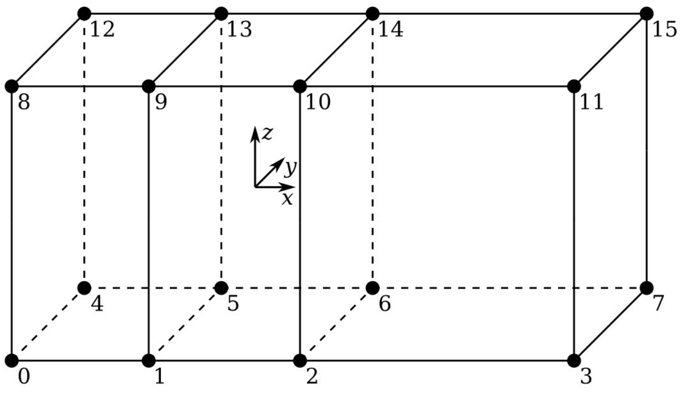

Figure 4-6: Graded control volume scheme

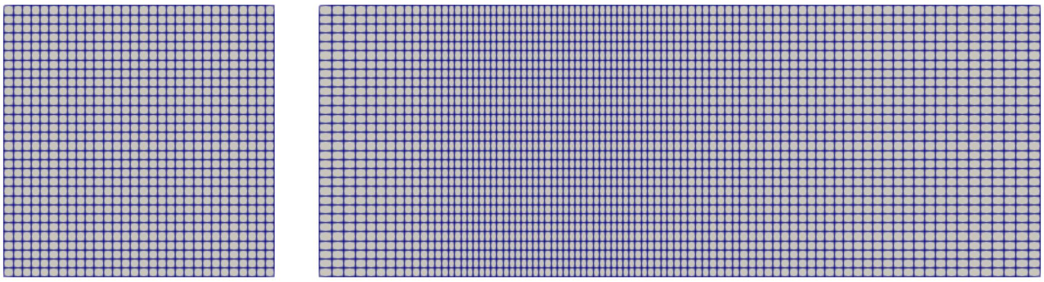

Figure 4-7: Graded control volume mesh

Dictionary file is listed below.

```
/*--------------------------------*- C++ -*----------------------------------*\
| =========                 |                                                 |
| \\      /  F ield         | OpenFOAM: The Open Source CFD Toolbox           |
|  \\    /   O peration     | Version:  6                                     |
|   \\  /    A nd           | Web:      www.OpenFOAM.org                      |
|    \\/     M anipulation  |                                                 |
\*---------------------------------------------------------------------------*/
FoamFile
{
    version     2.0;
    format      ascii;
    class       dictionary;
    object      blockMeshDict;
}
// * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * //

cv_x_min -20.0;
cv_x_max  60.0;
cv_y_min -15.0;
cv_y_max  15.0;
cv_z_min -15.0;
cv_z_max  15.0;
s2_x_min  -5.0;
s2_x_max  15.0;

convertToMeters 1.0;

vertices
(
    ( $cv_x_min $cv_y_min $cv_z_min )
    ( $s2_x_min $cv_y_min $cv_z_min )
    ( $s2_x_max $cv_y_min $cv_z_min )
    ( $cv_x_max $cv_y_min $cv_z_min )
    ( $cv_x_min $cv_y_max $cv_z_min )
    ( $s2_x_min $cv_y_max $cv_z_min )
    ( $s2_x_max $cv_y_max $cv_z_min )
    ( $cv_x_max $cv_y_max $cv_z_min )
    ( $cv_x_min $cv_y_min $cv_z_max )
    ( $s2_x_min $cv_y_min $cv_z_max )
    ( $s2_x_max $cv_y_min $cv_z_max )
    ( $cv_x_max $cv_y_min $cv_z_max )
    ( $cv_x_min $cv_y_max $cv_z_max )
    ( $s2_x_min $cv_y_max $cv_z_max )
```

```
    ( $s2_x_max $cv_y_max $cv_z_max )
    ( $cv_x_max $cv_y_max $cv_z_max )
);

blocks
(
    hex ( 0  1  5  4  8  9 13 12 ) ( 15 30 30 ) simpleGrading ( 0.5 1.0 1.0 )
    hex ( 1  2  6  5  9 10 14 13 ) ( 30 30 30 ) simpleGrading ( 1.0 1.0 1.0 )
    hex ( 2  3  7  6 10 11 15 14 ) ( 45 30 30 ) simpleGrading ( 2.0 1.0 1.0 )
);

edges
(
);

boundary
(
    inlet
    {
        type patch;
        faces
        (
            ( 0 8 12 4 )
        );
    }

    outlet
    {
        type patch;
        faces
        (
            ( 3 7 15 11 )
        );
    }

    walls
    {
        type patch;
        faces
        (
            // left
            (  0  1  9  8 )
            (  1  2 10  9 )
            (  2  3 11 10 )
            // right
            (  4 12 13  5 )
            (  5 13 14  6 )
            (  6 14 15  7 )
            // bottom
            (  0  4  5  1 )
            (  1  5  6  2 )
            (  2  6  7  3 )
            // top
            (  8  9 13 12 )
            (  9 10 14 13 )
            ( 10 11 15 14 )
        );
    }
);
```

```
mergePatchPairs
(
);
```

Mesh – Complex Grading

Scheme of complex control volume mesh grading is presented below.

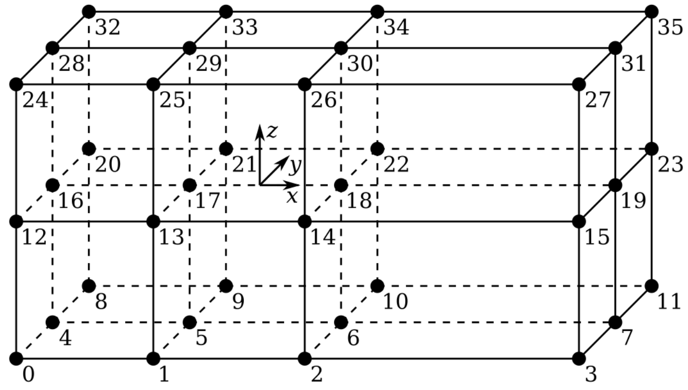

Figure 4-8: Complex graded control volume scheme

For symmetric cases only half of the geometry can be considered to reduce computation time.

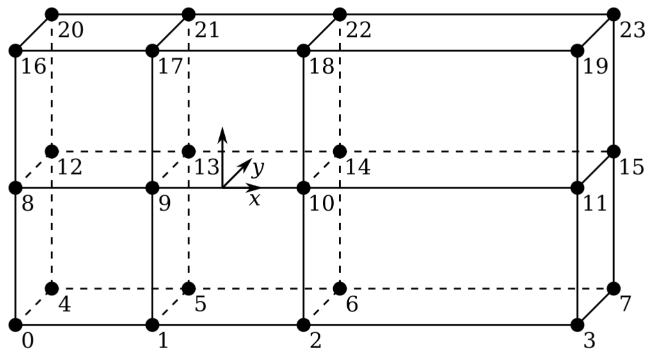

Figure 4-9: Symmetric control volume scheme

Dictionary file is listed below.

```
/*--------------------------------*- C++ -*----------------------------------*\
| =========                 |                                                 |
| \\      /  F ield         | OpenFOAM: The Open Source CFD Toolbox           |
|  \\    /   O peration     | Version:  6                                     |
|   \\  /    A nd           | Web:      www.OpenFOAM.org                      |
|    \\/     M anipulation  |                                                 |
\*---------------------------------------------------------------------------*/
```

```
FoamFile
{
    version     2.0;
    format      ascii;
    class       dictionary;
    object      blockMeshDict;
}
// * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * * //

cv_x_min -20.0;
cv_x_max  60.0;
cv_y_min -15.0;
cv_y_max  15.0;
cv_z_min -15.0;
cv_z_max  15.0;

s2_x_min  -5.0;
s2_x_max  15.0;

convertToMeters 1.0;

vertices
(
    ( $cv_x_min       0.0 $cv_z_min )
    ( $s2_x_min       0.0 $cv_z_min )
    ( $s2_x_max       0.0 $cv_z_min )
    ( $cv_x_max       0.0 $cv_z_min )
    ( $cv_x_min $cv_y_max $cv_z_min )
    ( $s2_x_min $cv_y_max $cv_z_min )
    ( $s2_x_max $cv_y_max $cv_z_min )
    ( $cv_x_max $cv_y_max $cv_z_min )
    ( $cv_x_min       0.0       0.0 )
    ( $s2_x_min       0.0       0.0 )
    ( $s2_x_max       0.0       0.0 )
    ( $cv_x_max       0.0       0.0 )
    ( $cv_x_min $cv_y_max       0.0 )
    ( $s2_x_min $cv_y_max       0.0 )
    ( $s2_x_max $cv_y_max       0.0 )
    ( $cv_x_max $cv_y_max       0.0 )
    ( $cv_x_min       0.0 $cv_z_max )
    ( $s2_x_min       0.0 $cv_z_max )
    ( $s2_x_max       0.0 $cv_z_max )
    ( $cv_x_max       0.0 $cv_z_max )
    ( $cv_x_min $cv_y_max $cv_z_max )
    ( $s2_x_min $cv_y_max $cv_z_max )
    ( $s2_x_max $cv_y_max $cv_z_max )
    ( $cv_x_max $cv_y_max $cv_z_max )
);

blocks
(
    hex (  0  1  5  4  8  9 13 12 ) ( 15 15 15 ) simpleGrading ( 0.5 2.0 0.5 )
    hex (  8  9 13 12 16 17 21 20 ) ( 15 15 15 ) simpleGrading ( 0.5 2.0 2.0 )
    hex (  1  2  6  5  9 10 14 13 ) ( 30 15 15 ) simpleGrading ( 1.0 2.0 0.5 )
    hex (  9 10 14 13 17 18 22 21 ) ( 30 15 15 ) simpleGrading ( 1.0 2.0 2.0 )
    hex (  2  3  7  6 10 11 15 14 ) ( 45 15 15 ) simpleGrading ( 2.0 2.0 0.5 )
    hex ( 10 11 15 14 18 19 23 22 ) ( 45 15 15 ) simpleGrading ( 2.0 2.0 2.0 )
);
```

```
edges
(
);

boundary
(
    inlet
    {
        type patch;
        faces
        (
            (  0  8 12  4 )
            (  8 16 20 12 )
        );
    }

    outlet
    {
        type patch;
        faces
        (
            (  3  7 15 11 )
            ( 11 15 23 19 )
        );
    }

    sym
    {
        type symmetryPlane;
        faces
        (
            // left
            (  0  1  9  8 )
            (  8  9 17 16 )
            (  1  2 10  9 )
            (  9 10 18 17 )
            (  2  3 11 10 )
            ( 10 11 19 18 )
        );
    }

    walls
    {
        type patch;
        faces
        (
            // right
            (  4 12 13  5 )
            ( 12 20 21 13 )
            (  5 13 14  6 )
            ( 13 21 22 14 )
            (  6 14 15  7 )
            ( 14 22 23 15 )

            // bottom
            (  0  4  5  1 )
            (  1  5  6  2 )
            (  2  6  7  3 )
```

```
            // top
            ( 16 17 21 20 )
            ( 17 18 22 21 )
            ( 18 19 23 22 )
        );
    }
);

mergePatchPairs
(
);
```

Mesh – Aircraft Geometry Model

OpenVSP can be used to create aircraft 3D model and export it to `STL`. OpenFOAM `snappyHexMesh` utility is used to combine control volume mesh with aircraft geometry model. Volume near aircraft geometry can be refined due to achieve better results. Meshing parameters are specified in `system/snappyHexMeshDict` file.

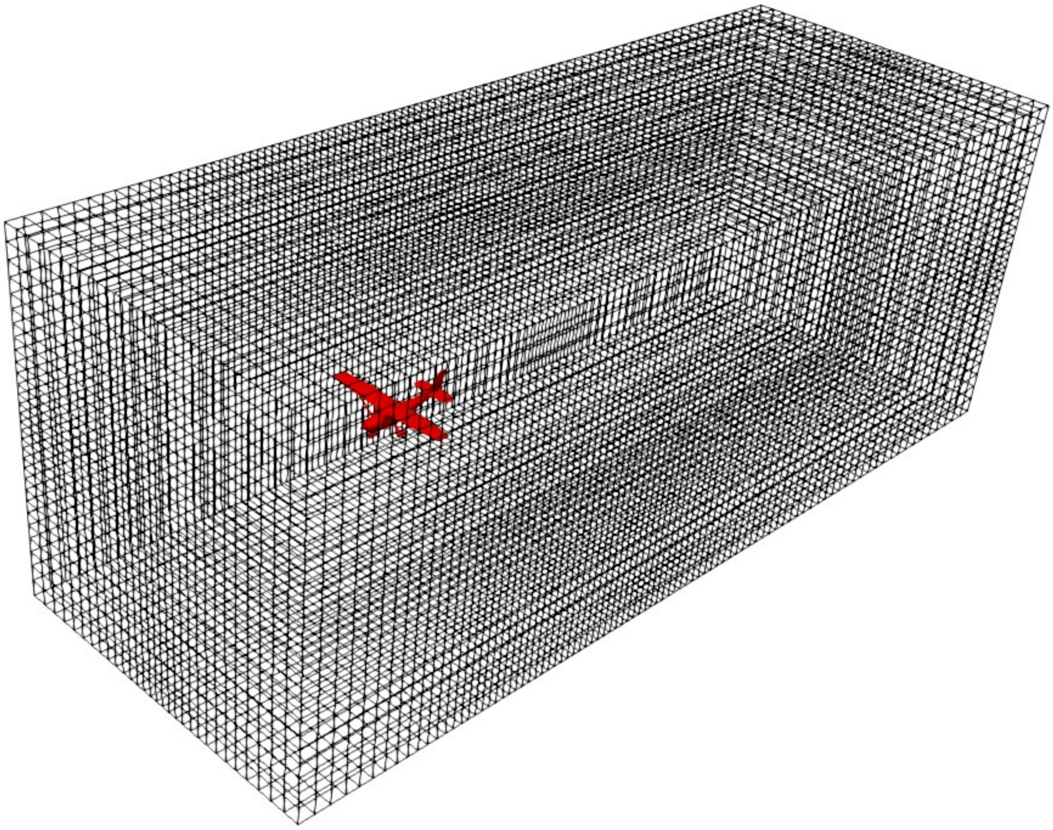

Figure 4-10: OpenFOAM complete mesh

Initial and Boundary Conditions

The inlet boundary conditions are given as follows. [14]

$$k=\frac{3}{2}(IV)^2 \tag{4.1}$$

$$\omega=\frac{k^{0.5}}{C_\mu L} \tag{4.2}$$

where: [12], [13], [15], [16], [17]

$$I=0.16\,\mathrm{Re}^{-1/8} \tag{4.3}$$

$$C_\mu=0.09 \tag{4.4}$$

Control

Notice that, as this is a steady-state simulation, time should be treated rather as iteration parameter rather than as a physical time. Maximum number of iterations in set `system/controlDict` dictionary file, `endTime` is a maximum number of iterations when `deltaT` is 1.

Solution

Case solving is terminated when pressure, velocity, turbulent kinetic energy and specific turbulence dissipation rate initial residual of the field equations falls below threshold values defined in `residualControl` dictionary in `system/fvSolution` file.

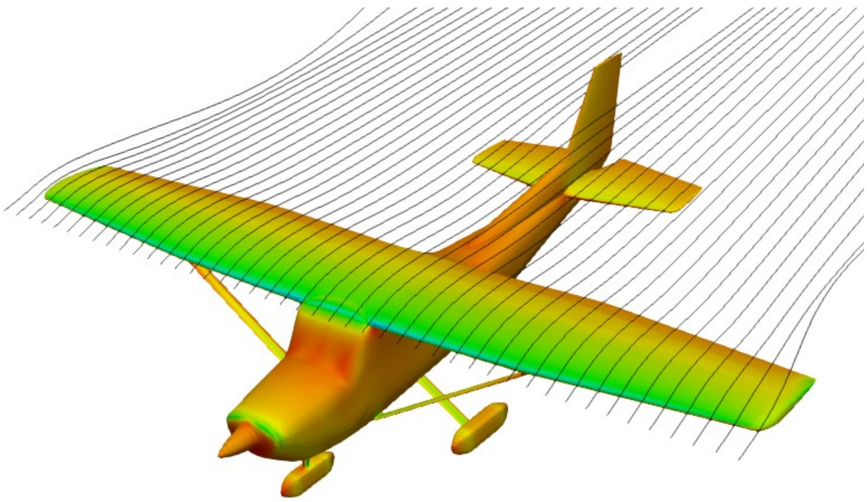

Figure 4-11: Streamlines and kinematic pressure distribution

If divergent or unstable behavior is observed decreasing under-relaxation factors defined in `relaxationFactors` dictionary in `system/fvSolution` file may improve solution convergence.

Physical quantity	**Under-Relaxation Factor**
Kinematic pressure	0.2 – 0.3
Velocity	0.5 – 0.7
Turbulent kinetic energy	0.5 – 0.7
Turbulence dissipation rate	0.5 – 0.7

Table 4-1: Commonly used under-relaxation factors [17], [18]

4.3. Results

4.3.1. XFOIL

XFOIL computations results, compared to the data available in [5] and [6], are shown in the following figures.

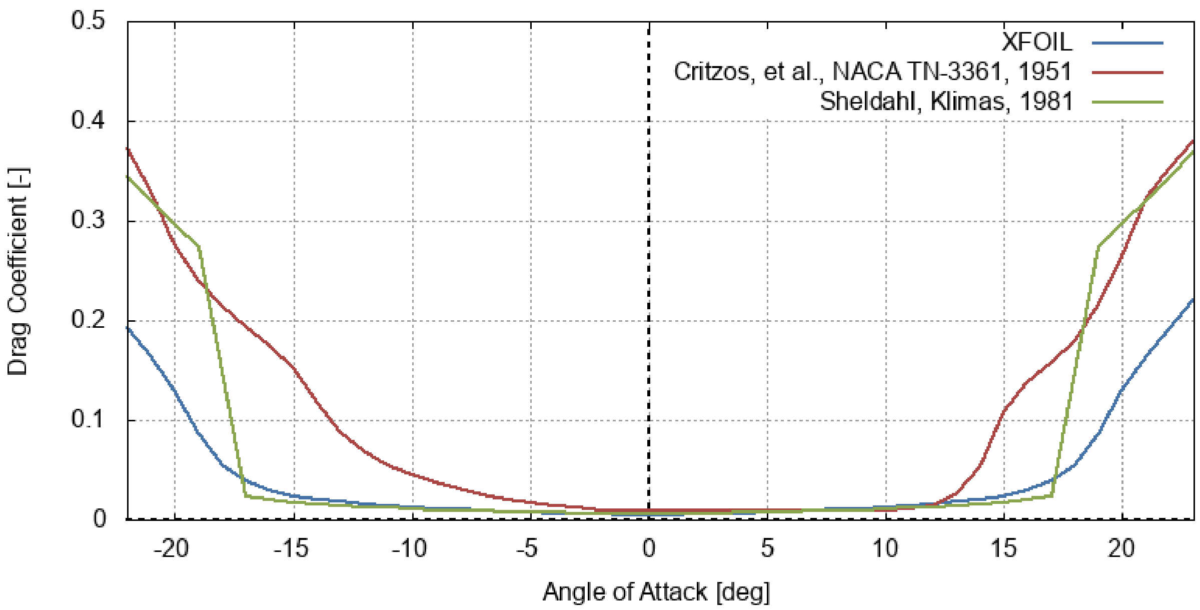

Figure 4-12: NACA 0012 airfoil drag coefficient

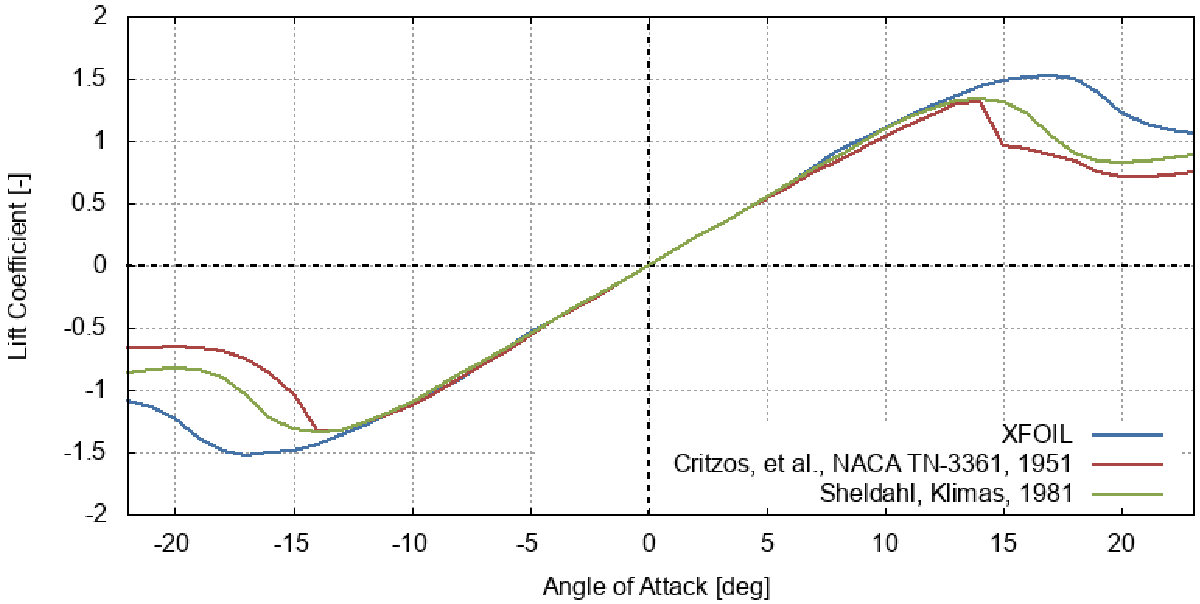

Figure 4-13: NACA 0012 airfoil lift coefficient

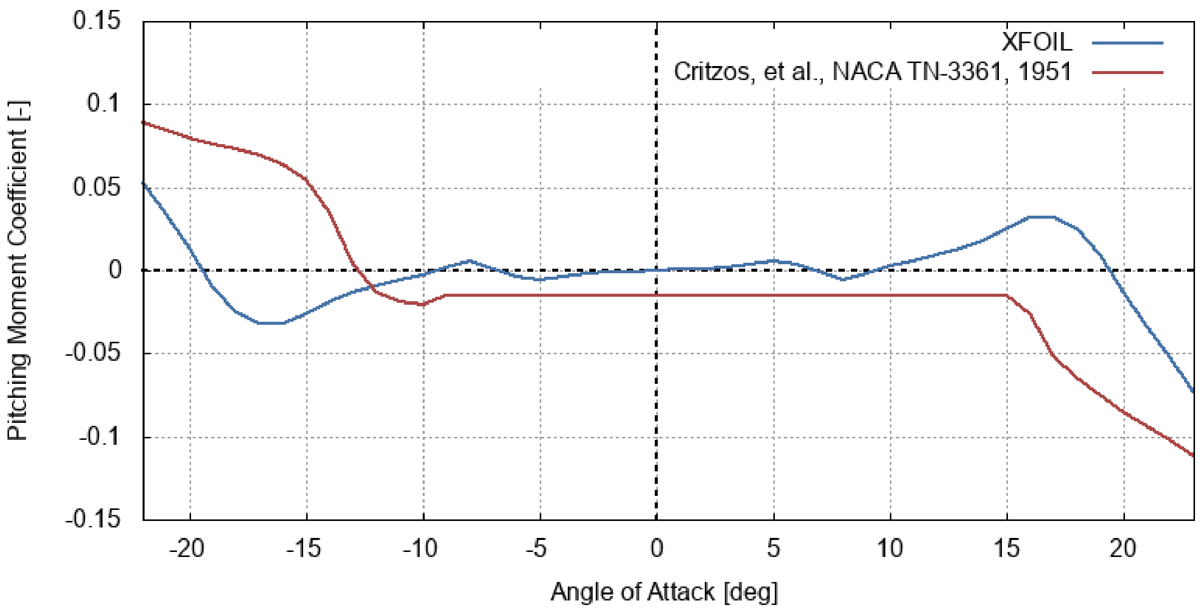

Figure 4-14: NACA 0012 airfoil pitching moment coefficient

4.3.2. VSPAERO

VSPAERO computations results, compared to the OpenFOAM computations results, are shown in the following figures.

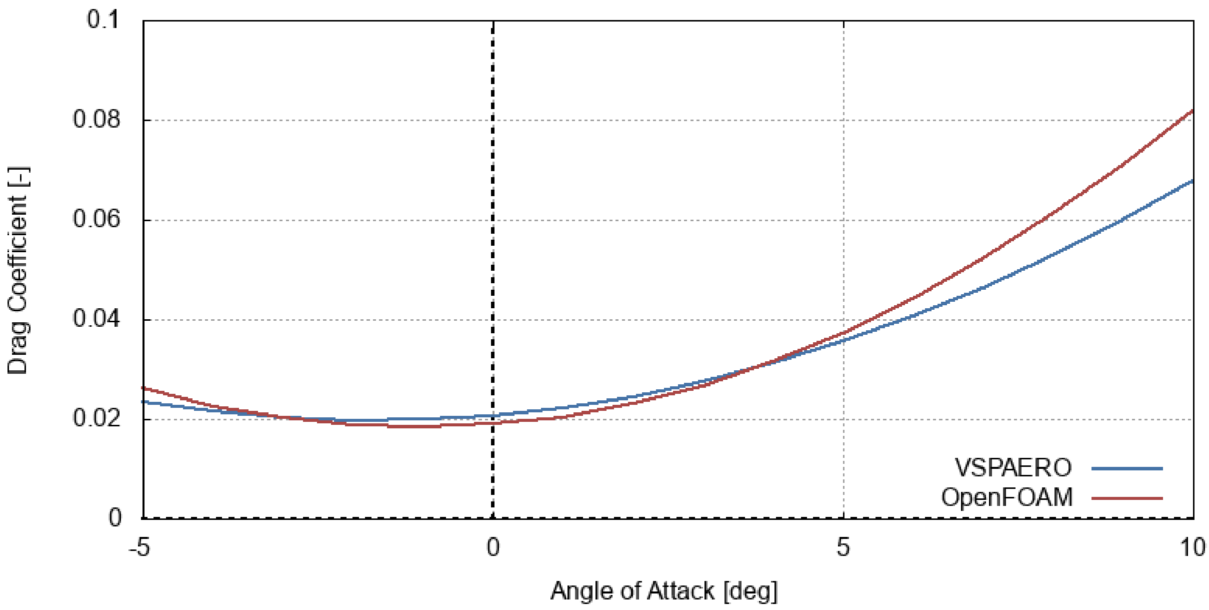

Figure 4-15: Drag coefficient

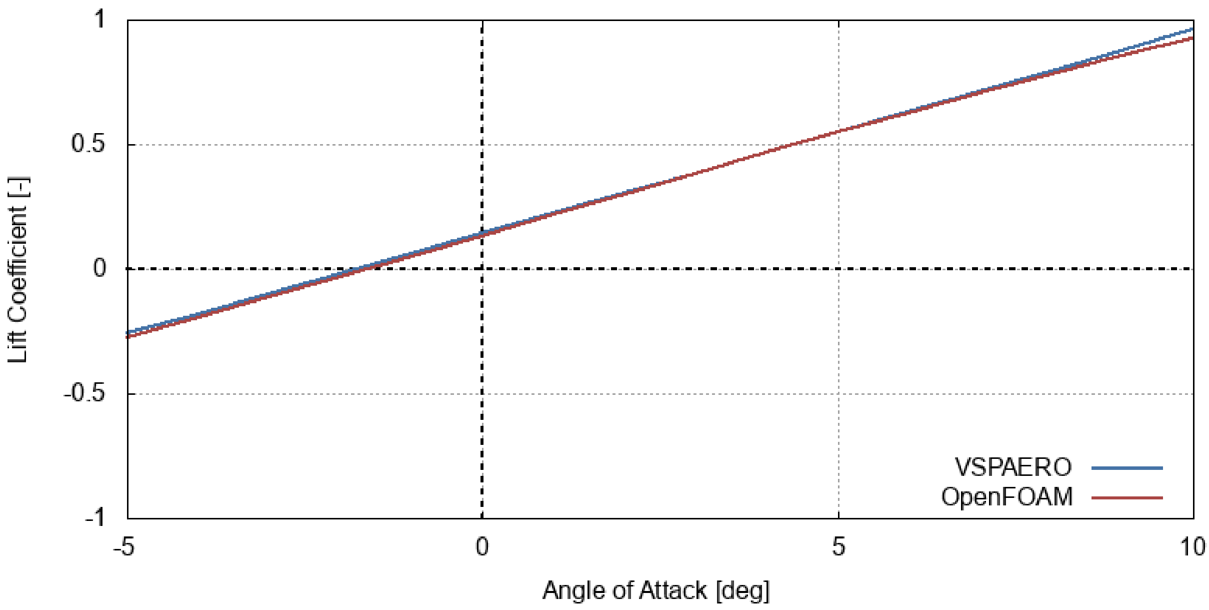

Figure 4-16: Lift coefficient

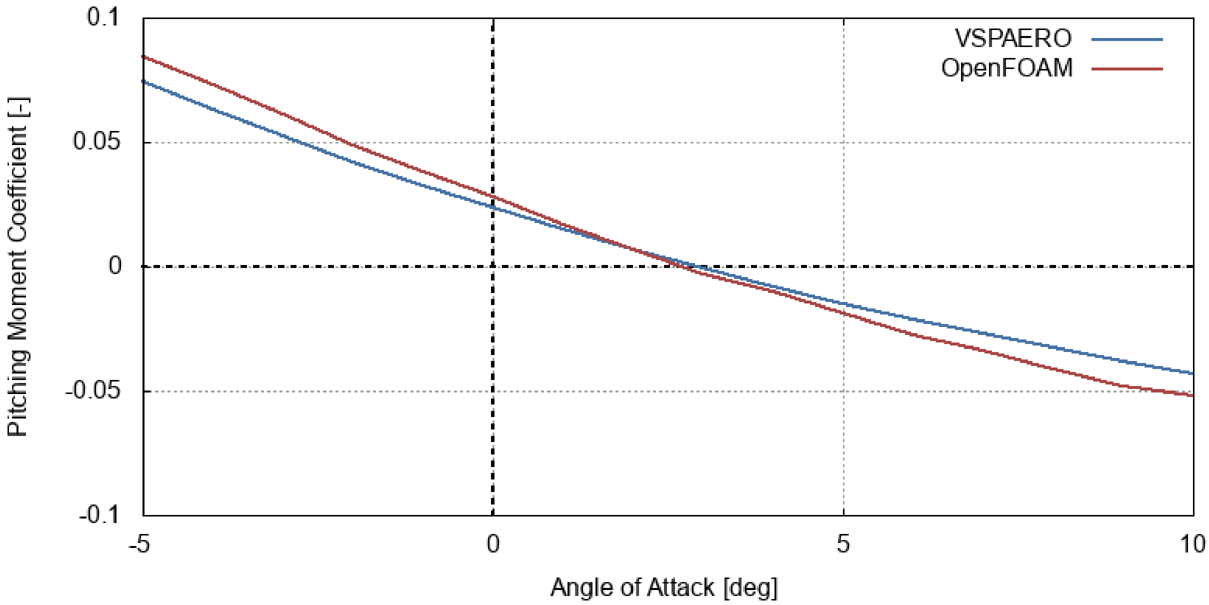

Figure 4-17: Pitching moment coefficient

4.3.3. OpenFOAM

OpenFOAM computations results, compared to the data available in [19], are shown in the following figures.

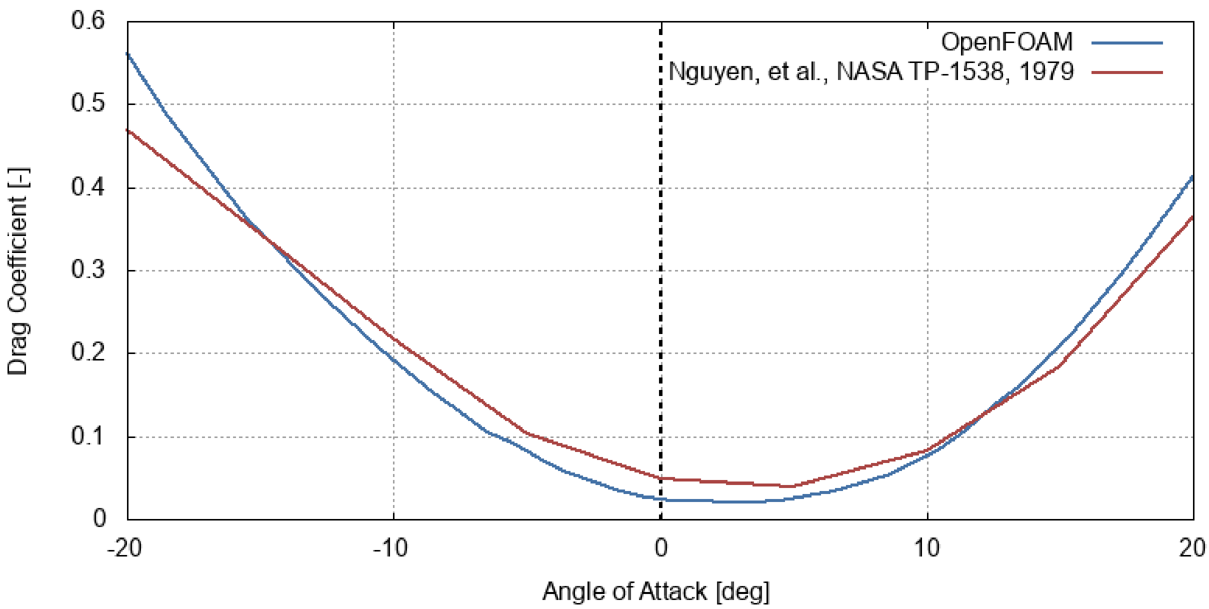

Figure 4-18: F-16 drag coefficient

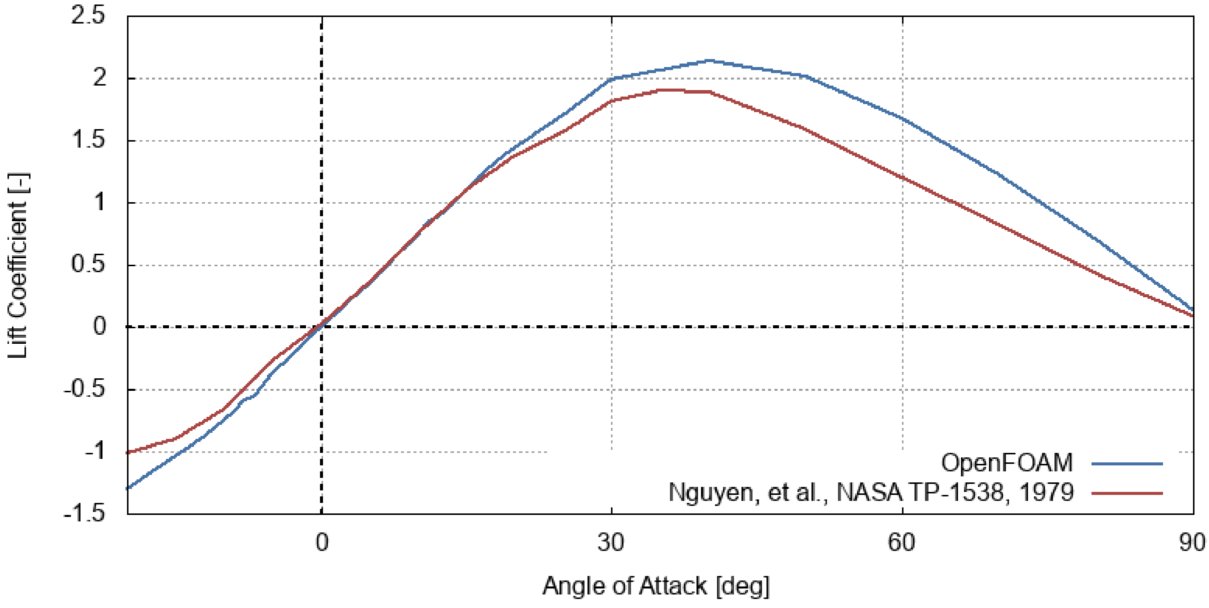

Figure 4-19: F-16 lift coefficient

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