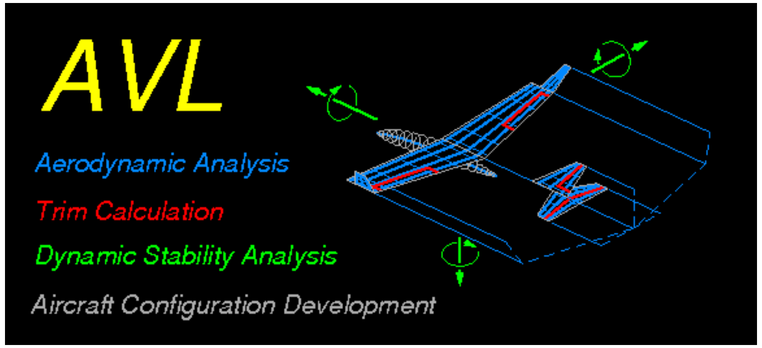
Athena Vortex Lattice (AVL)

- Open-Source code developed in MIT by Prof Mark Drela's lab
  - Aerodynamic and flight dynamic analysis
  - Rigid body aircraft of arbitrary configuration



- Analysis method
  - Extended vortex lattice model for the lifting surfaces
  - Slender-body model for fuselages and nacelles
  - General nonlinear flight states can be specified
- The flight dynamic analysis combines a full linearization of the aerodynamic model about any flight state, together with specified mass properties

http://web.mit.edu/drela/Public/web/avl/

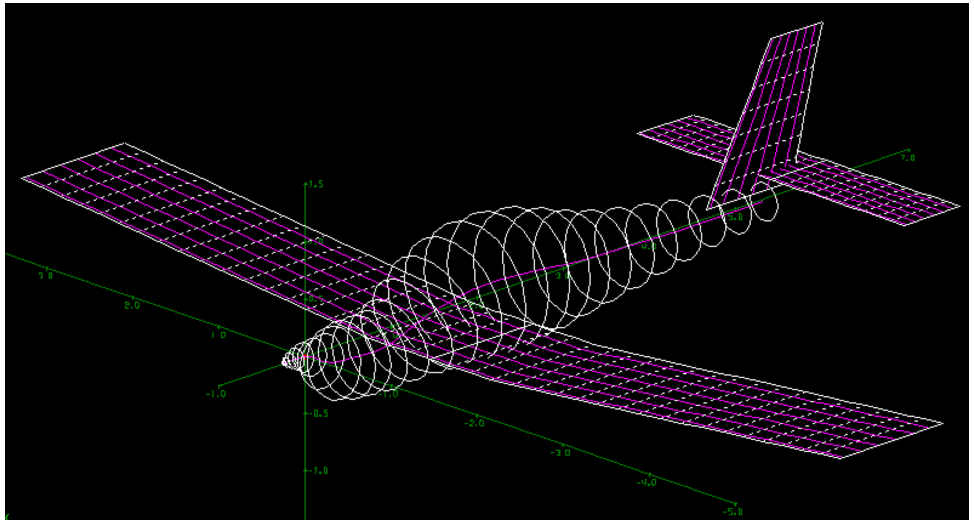
AVL Features

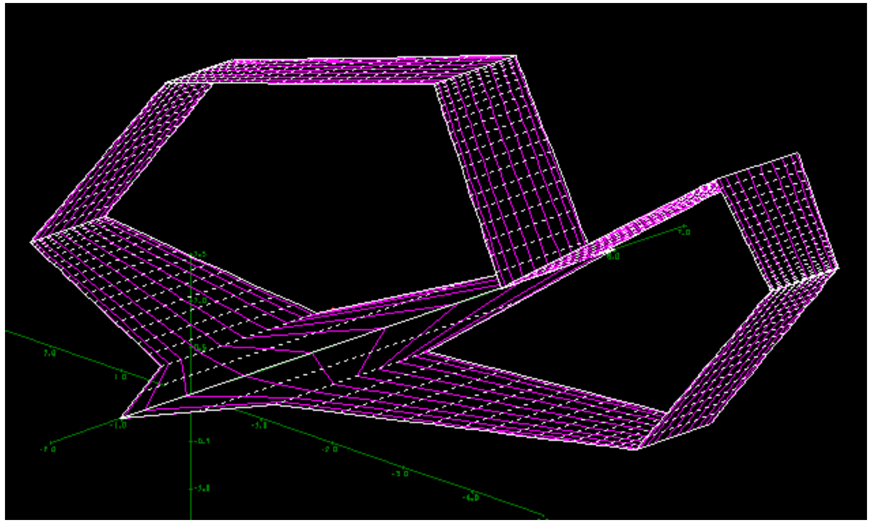
- Keyword driven geometry input file (similar to XFOIL)
- Control surface deflections
- Trim calculation
- Eigenmode analysis
- Optional mass definition file (for trim analysis, eigenmode analysis)

- Input files:
  - xxx.avl – required main input file defining the configuration geometry
  - xxx.mass – optional file giving masses and inertias, and dimensional units
  - xxx.run – optional file defining the parameter for some number of run cases
  - Airfoil, fuselage profile coordinates

Fuselage Analysis Limitations

- 3D Panel method
  - Works only for streamlined fuselage
  - Approximation of elliptic cross section
- Model fuselage as a wing
  - Create “wings” in symmetry and/or horizontal plane
  - This “wing” will be used to predict effect of fuselage on lift, pitch, side force, etc.
- Fuselage with non-streamlined shape may fail to analyze. Better to skip them





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Input Files: Airfoil, Fuselage profile

- Airfoil and fuselage profile files have standard airfoil file format:

- XY coordinates starting from trailing edge (tail), upper profile points, leading edge, Trailing edge(tail) of the lower profile.

V05-8_fuseSide.dat | V05-8_hStab.dat

```
+1.0000   +0.0013
+0.9965   +0.0018
...
+0.0003   +0.0031
+0.0000   +0.0000
+0.0003   -0.0031
...
+0.9965   -0.0018
+1.0000   -0.0013
```

Input file: main input (*.avl)

- Show sample.avl

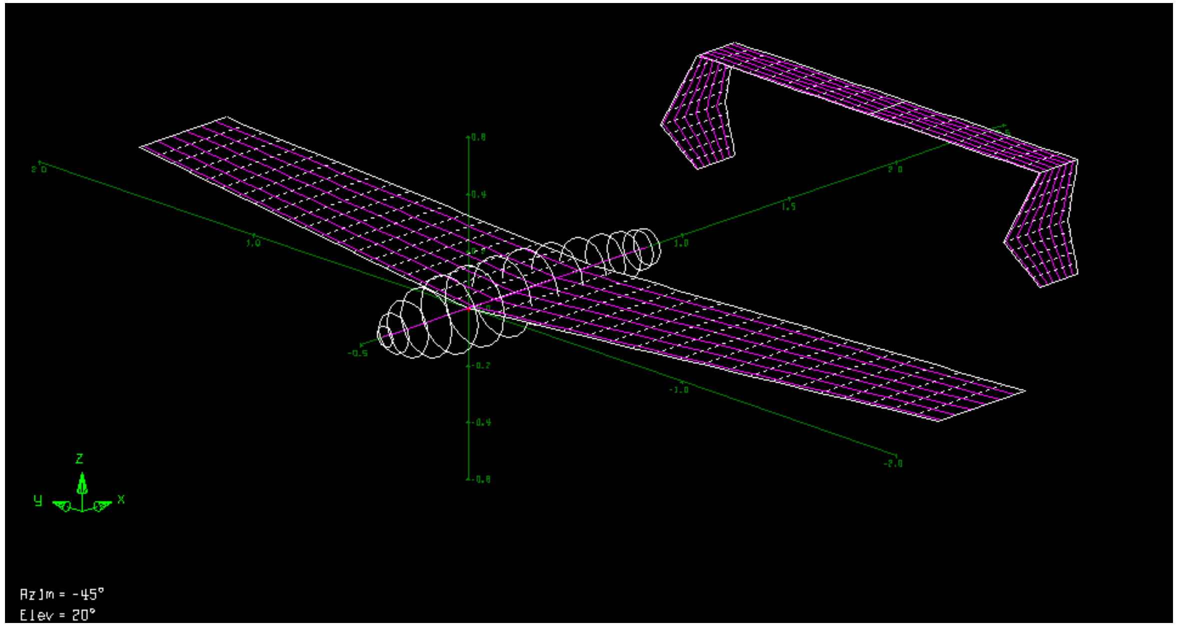
testFile.avl

```
AVL INPUT FILE - GENERATED BY AERODYNAMICS.PY
0
0 0 0
11.400000 1.200000 9.500000
0 0 0
0
#====================BODY======================#
BODY
FUSELAGE
28 2
BFIL
testFile_fus.dat
#==================MAIN WING===================#
SURFACE
WING
6 0 20 0
YDUPLICATE
0
ANGLE
2.000000
```

http://web.mit.edu/drela/Public/web/avl/

Running AVL

- Load configuration file
- Display aircraft (optional)
- Define operating conditions
- Analysis
    - Single point
    - Multiple points
    - Trim analysis



AVL Example

- Run avl.exe
- Load configuration and display
  - `load sample.avl`
  - `oper` – enter operation runcase setup
  - `G` – plot geometry. Check if the input file you created is correct
  - `(enter)` to exit plot geometry mode

- Modify parameters
  - `M` – enter parameter change mode
  - Type required parameter name
  - `V(enter) 20` – velocity
  - `M(enter) 25` – mass
  - `G(enter) 9.8` – gravity acceleration
  - `(enter)` exit parameter modification mode

Input command

```
Load sample.avl
Oper
G
M
V 20
M 25
G 9.8
```

AVL Example

- Single point analysis
  - After all files are loaded and parameters entered, press **x** to run analysis

- Display results
  - Type required option

```
ST   stability derivatives     FT   total   forces
SB   body-axis derivatives     FN   surface forces
RE   reference quantities      FS   strip   forces
                               FE   element forces
DE   design changes            VM   strip shear,moment
O ptions                       HM   hinge moments
```

  - Display on screen **(enter)**
  - Or save to file **filename (enter)**

Input command

```
X
St stab_result.txt
HM hinge_result.txt
RE ref_values.txt
```

```
---------------------------------------------------------------
Vortex Lattice Output -- Total Forces

Configuration: Hybrid drone
    # Surfaces =   6
    # Strips   =  80
    # Vortices = 480

 Sref =  1.6502       Cref = 0.50180       Bref =  3.7252
 Xref = 0.28500       Yref =  0.0000       Zref =  0.0000

Standard axis orientation,  X fwd, Z down

Run case:  -unnamed-

 Alpha =   0.00000     pb/2V =   0.00000     p'b/2V =   0.00000
 Beta  =   0.00000     qc/2V =   0.00000
 Mach  =     0.000     rb/2V =   0.00000     r'b/2V =   0.00000

 CXtot =  -0.03342     Cltot =   0.00000     Cl'tot =   0.00000
 CYtot =   0.00000     Cmtot =   0.01965
 CZtot =  -0.30208     Cntot =   0.00000     Cn'tot =   0.00000

 CLtot =   0.30208
 CDtot =   0.03342
 CDvis =   0.03000     CDind =   0.00342
 CLff  =   0.29514     CDff  =   0.00339    | Trefftz
 CYff  =   0.00000         e =    0.9722    | Plane

  AILERON         =   0.00000
  ELEVATOR        =   0.00000

---------------------------------------------------------------
```

```
---------------------------------------------------------------

Stability-axis derivatives...

                             alpha                beta
                  ----------------    ----------------
z' force CL |    CLa =   7.262788    CLb =   0.000000
y  force CY |    CYa =   0.000000    CYb =  -0.340930
x' mom.  Cl'|    Cla =   0.000000    Clb =  -0.028129
y  mom.  Cm |    Cma =  -0.498287    Cmb =   0.000000
z' mom.  Cn'|    Cna =   0.000000    Cnb =   0.087912

                     roll rate  p'      pitch rate  q'        yaw rate  r'
                  ----------------    ----------------    ----------------
z' force CL |    CLp =   0.000000    CLq =  11.328282    CLr =   0.000000
y  force CY |    CYp =   0.064554    CYq =   0.000000    CYr =   0.198301
x' mom.  Cl'|    Clp =  -0.578325    Clq =   0.000000    Clr =   0.072146
y  mom.  Cm |    Cmp =   0.000000    Cmq = -19.250702    Cmr =   0.000000
z' mom.  Cn'|    Cnp =  -0.027299    Cnq =   0.000000    Cnr =  -0.112051

                  AILERON      d1     ELEVATOR     d2
                  ----------------    ----------------
z' force CL |   CLd1 =   0.000000   CLd2 =   0.012737
y  force CY |   CYd1 =  -0.000028   CYd2 =   0.000000
x' mom.  Cl'|   Cld1 =  -0.006470   Cld2 =   0.000000
y  mom.  Cm |   Cmd1 =   0.000000   Cmd2 =  -0.039289
z' mom.  Cn'|   Cnd1 =   0.000173   Cnd2 =   0.000000
Trefftz drag| CDffd1 =   0.000000 CDffd2 =   0.000171
span eff.   |    ed1 =   0.000000    ed2 =   0.034853



Neutral point  Xnp =   0.319428

Clb Cnr / Clr Cnb  =   0.496944    (  > 1 if spirally stable )
```

Trim analysis

- Trim is the equilibrium condition

$L = W$ - Lift is equal to weight -> level flight

$M = 0$ - Pitch moment is zero -> no rotation about the CG

- Aircraft lift can be controlled by adjusting angle of attack
- Aircraft pitch moment is controlled by elevator

$$C_{L_{req}} = \frac{2W}{\rho V^2 S_{ref}}$$

$$C_{L_0} + C_{L_\alpha}\alpha + C_{L_{\delta e}}\delta_e = C_{L_{req}}$$

$$C_{m_0} + C_{m_\alpha}\alpha + C_{m_{\delta e}}\delta_e = 0$$

AVL Example: trim analysis

- Find α, δ_e
- Such that $C_L = C_{L_{req}}$
  $C_m = 0$

- Enter angle of attack constraint
  - When in operating mode press **A** to setup angle of attack constraint
  - **`C 0.2568`** - enter required lift coefficient
- Enter elevator deflection constraint
  - **`D1`** – select elevator number (depends on input avl file)
  - **`PM 0`** – specify required pitch moment is equal to zero
- Enter **`x`** – to execute trim analysis
- Display results
  - Calculated angle of attack and elevator is the result of trim analysis

Run AVL: analyze aircraft

- Analyze light aircraft aerodynamics and S&C

#	cmd	description
1	`load testFile.avl`	load configuration file "testFile.avl"
2	`oper`	compute operating point run cases
3	`m`	modify parameters (parameters shown in next pag e)
4	`d4`	select control surface #4 (elevator)
5	`pm 0`	required pitching moment = 0 (elevator deflection will be set to obtain $C_m = 0$
6	`x`	execute runcase (analysis)
7	`st`	display (or save to file) stability derivatives
8	`ft`	display (or save to file) total forces
9	`quit`	quit the program