AE4233 Advanced design methods

MDO Practical session 3

Introduction to the Quasi-3D Aerodynamic Solver

Background theory and user manual of a rapid aerodynamic solver for 3D wings to be used in MDO assignment

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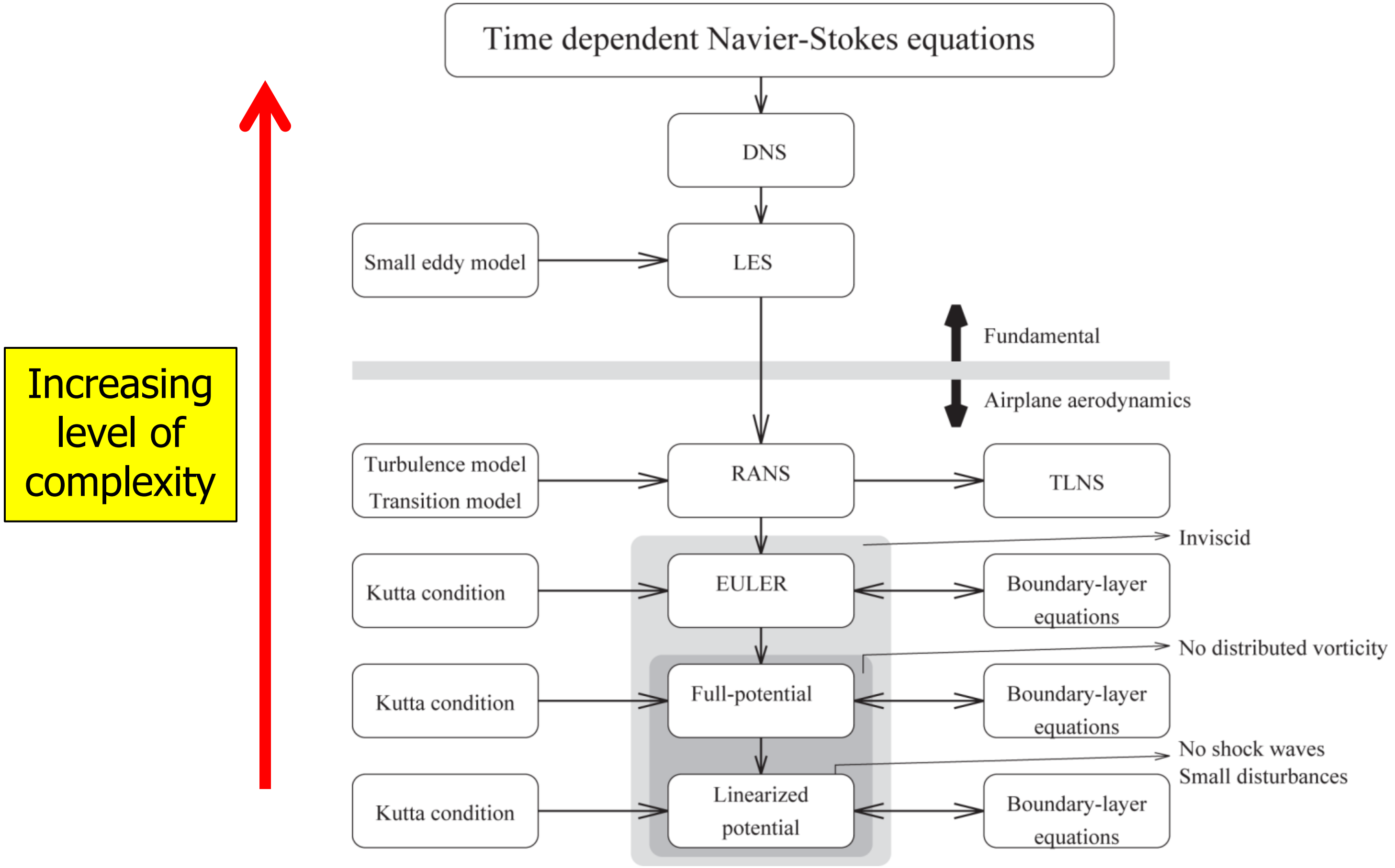
Outlines of this tutorial

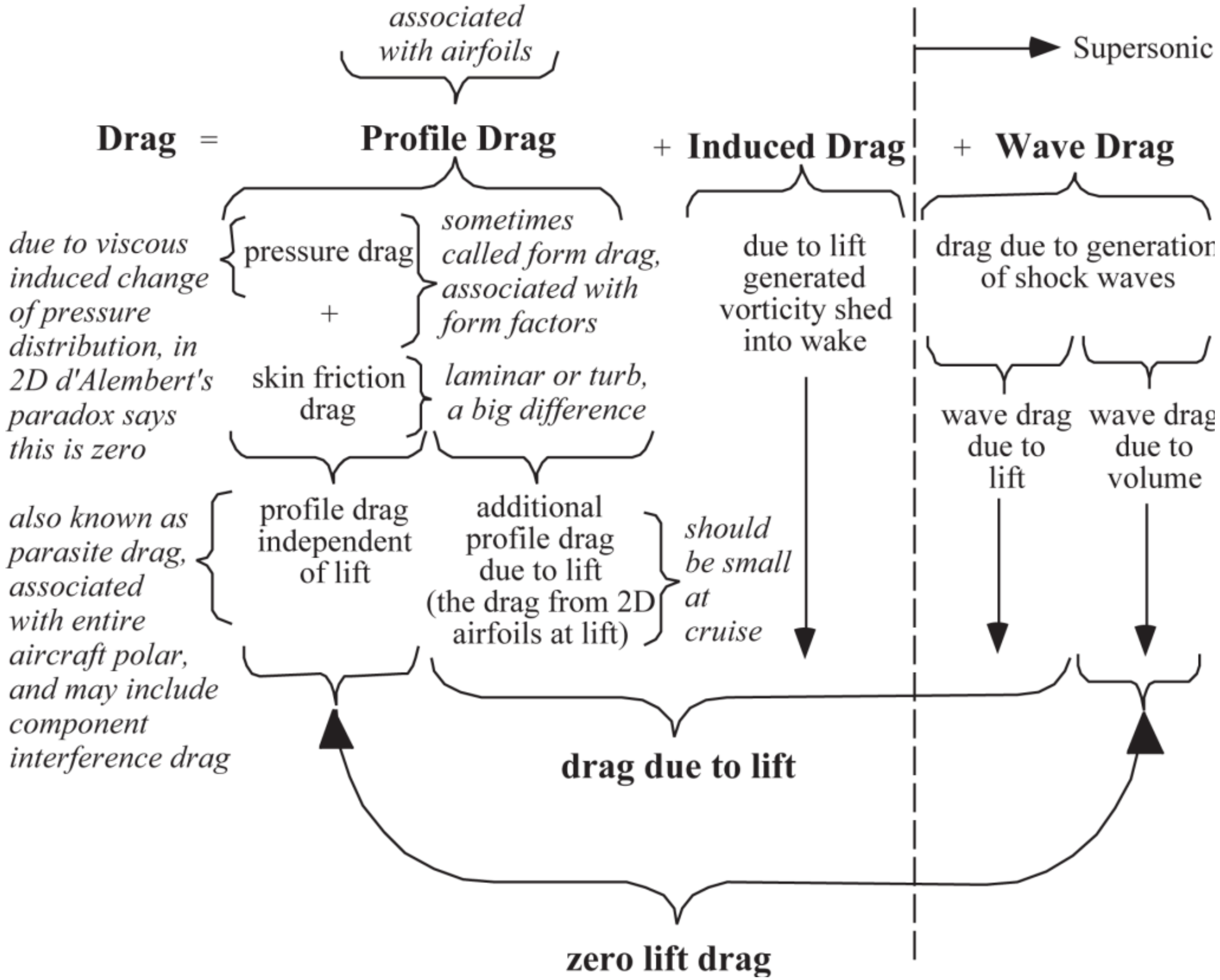
- Background theory on computational fluid dynamics
- Structure and functionality of the Q-3D aerodynamic solver

- Interactive tutorial on the use of Q-3D AeroSolver

Overview of methods for Aerodynamic flow analysis and modelling



Drag & its contributions



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Vortex Lattice Method (VLM)

- The VLM is based on the solution of the Laplace equation:
  - Singularities are placed on a **surface**.
  - The non-penetration condition is satisfied at a number of control points.
  - A system of linear algebraic equations is solved to determine singularity strengths.

Main Assumptions in VLM

- The wing thickness is ignored
- Boundary Conditions are applied on the wing mean surface (camber line in 2D), not the actual surface.
- Compressibility corrections are used to account for Mach effect.

ATT! VLM results are **not** valid for high Mach numbers due to the poor accuracy of compressibility correction!

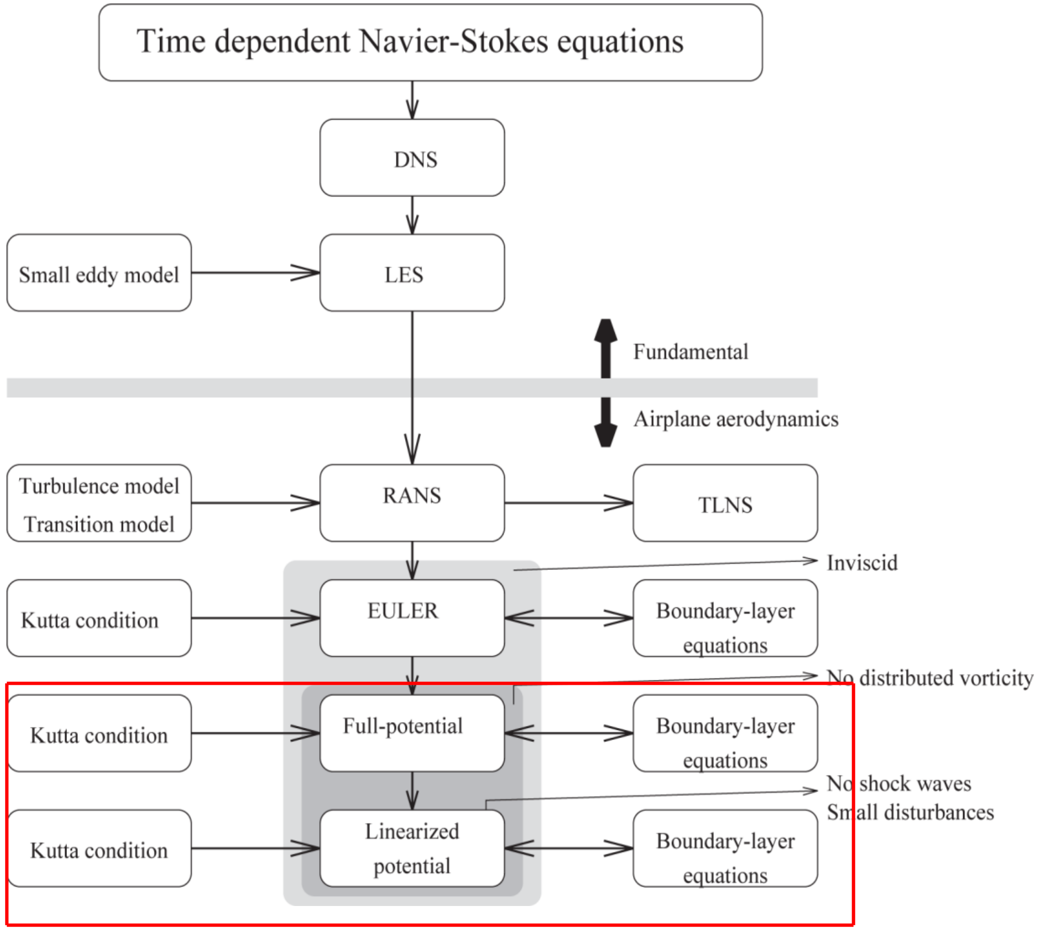
The Quasi-3D Aerodynamic Solver

Q-3D AeroSolver uses a combination of

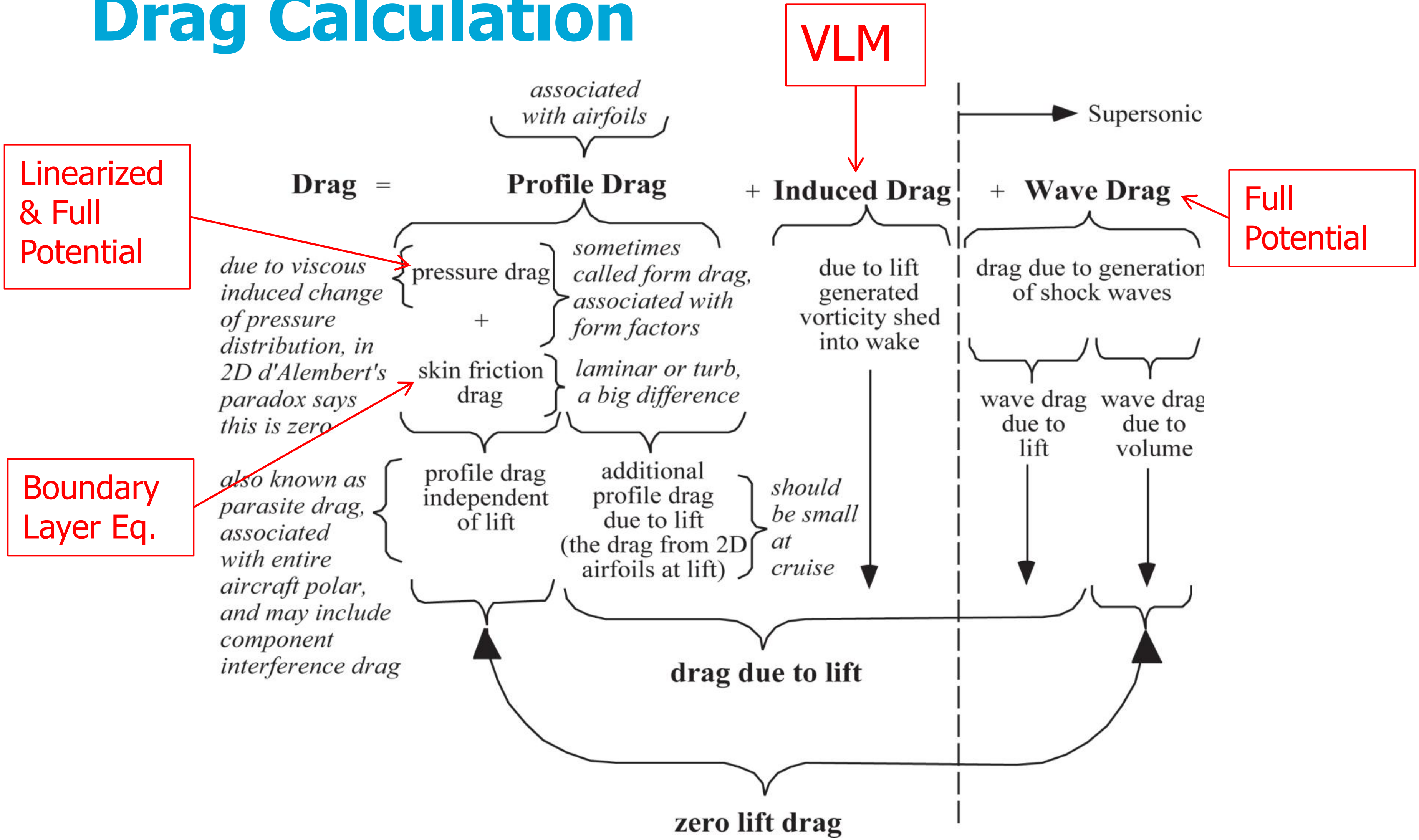
- *VLM*
- *Linearized Potential*
- *Full Potential*
- *and Boundary layer equations*

to calculate lift, drag and pitching moment of a 3D wing.

Q-3D AeroSolver is a Matlab application that integrates existing aerodynamic codes (Xfoil, AVL, VGK)



Drag Calculation



Quasi-3D AeroSolver makes use of…

…a VLM method to compute the **(3D) wing spanwise lift** distribution.

Integrated code: *AVL* (*Athena Vortex Lattice*)

To compute the **(2D) airfoil drag contribution**…

- If M<0,6 (low subsonic speed)

→Linearized Potential flow + BL

Integrated code: *Xfoil*

- If 0.6<M<1 (transonic speed)

→Full-Potential method + BL

Integrated code: *VGK (Viscous Garabedian and Korn)*

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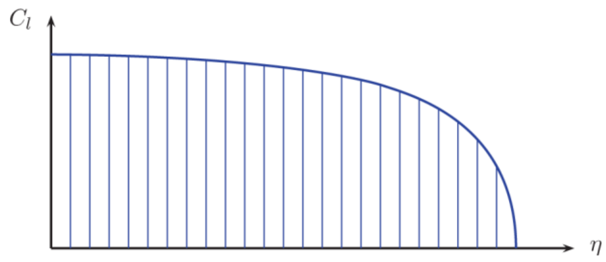
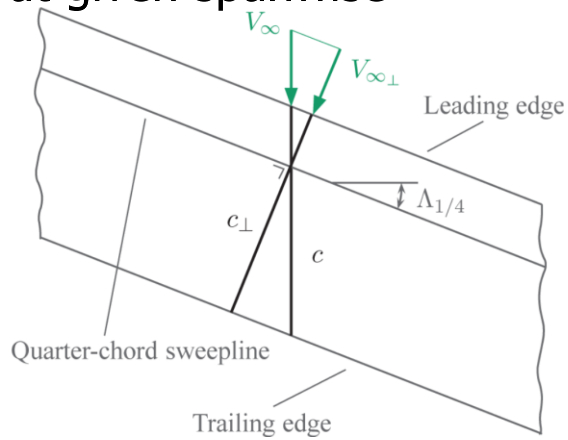
Wing Lift & Drag estimation using the Q3D AeroSolver:

In each analysis session, the following steps are performed:

Step 1. Execution of VLM code to find the spanwise lift distribution.



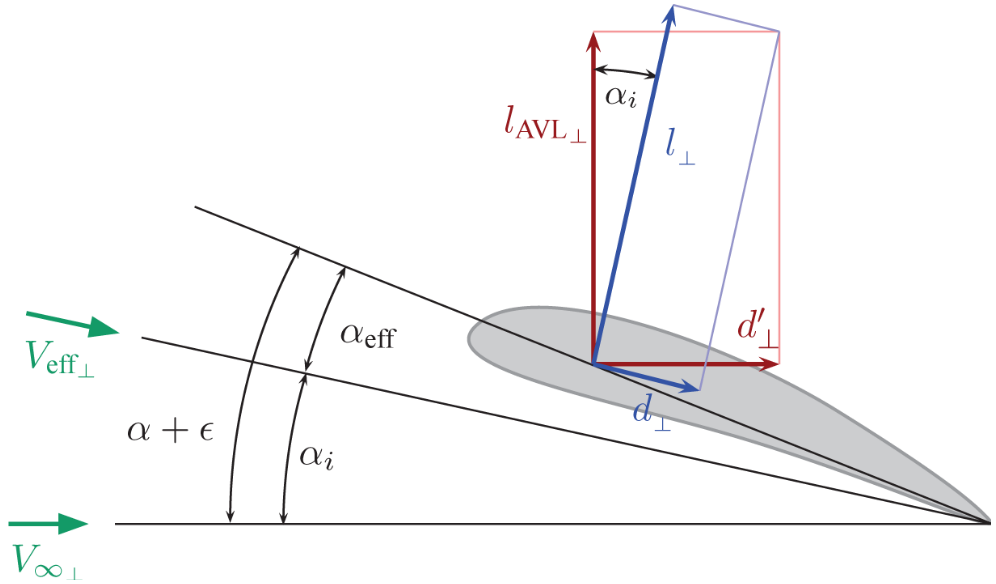
Step 2. Extraction of 2D airfoil sections (from the geometrical wing model) and local lift coefficients (from the spanwise lift distribution), at given spanwise stations.



Step 3. Application of sweep theory to translate streamwise properties (Cl, M, Re, V) to wing sections perpendicular to ¼ chord line.

Wing Lift & Drag estimation using Q3D AeroSolver (2):

Step 4. Iteration to find proper downwash angle (α_i) for the given Cl at each spanwise position using 2D solver.



Wing Lift & Drag estimation using Q3D AeroSolver (3):

Step 5. Determination of the drag coefficient of the selected wing sections.

Step 6. Translation of the perpendicular properties to streamwise using sweep theory.

Step 7. Calculation of the total wing drag coefficient (i.e., induced drag + profile drag + wave drag)

Flow Chart of Q-3D AeroSolver operations

Step 1

Run AVL to find lift distribution on wing

Step 2

Interpolation on the lift distribution to find lift coefficient at given spanwise airfoil section

$$C_{l_{\mathrm{AVL}}}$$

Step 3

Apply sweep theory to find local lift coefficient

$$C_{l_{\mathrm{AVL}_\perp}} = C_{l_{\mathrm{AVL}}} \sec^2 \Lambda$$

$$M_{\infty_\perp} = M_\infty \cos \Lambda$$

Step 4

Find induced angle of attack, α_i

1. Set $\alpha_i = 0$, $C_{d_\perp} = 0$ and $\alpha_{\mathrm{eff}} = \alpha + \epsilon - \alpha_i$
2. $C_{l_\perp} = \dfrac{C_{l_{\mathrm{AVL}_\perp}} \cos^2 \alpha_i + C_{d_\perp} \sin \alpha_i}{\cos \alpha_i}$
3. $V_{\mathrm{eff}}^{**} = \dfrac{V_\infty^{**}}{\cos \alpha_i}$ where $V_\infty^{**} = V_\infty \cos \Lambda$

   $$Re_{\mathrm{eff}} = \frac{V_{\mathrm{eff}}^{**}}{V_\infty^{**}} Re_\infty$$
4. Run Xfoil/MSES/VGK to find $C_{d_\perp}$ and α_{eff}

   $[C_{d_\perp}, \alpha_{\mathrm{eff}}]$ = Xfoil/VGK/MSES(Re_{eff}, $M_\perp$, $C_{l_\perp}$)
5. $\alpha_i = \alpha + \epsilon - \alpha_{\mathrm{eff}}$
6. Repeat steps 2-5 until α_i converged

Step 5

Drag coefficients in streamwise direction (V_∞)

$$C_{d'_{f_\perp}} = C_{d_{f_\perp}} \frac{\cos \alpha_i}{\cos^2 \alpha_i}$$

$$C_{d'_{p_\perp}} = C_{d_{p_\perp}} \frac{\cos \alpha_i}{\cos^2 \alpha_i}$$

* MSES & VGK report friction and pressure drag, Xfoil reports total drag $C_{d'_\perp}$

Step 6

Apply sweep theory to find local lift/drag coefficient

$$C_{d'} = C_{d'_{f_\perp}} + C_{d'_{p_\perp}} \cos^3 \Lambda$$

Step 7

Calculate 3D drag coefficient

$$C_D = \frac{2}{S_{\mathrm{ref}}} \int_0^{b/2} C_{d'} c \, dy + C_{D_i}$$

ATT! The analysis code to be used is automatically selected based on Mach number: Xfoil for M<0.6, VGK for M>0.6.

MSES is not installed in this version of Q3D Solver

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Using the Q3D AeroSolver code

- Download Q3D_Solver from blackboard. Open Start.m
- **Inputs:**
  - Wing planform geometry (coordinates of sections apex points, chord lengths, twist angle)
  - Wing shape (airfoil geometry expressed as CST coefficients)
  - Flight Condition (Mach, Re, altitude, speed, either angle of attack or lift coefficient)
  - Viscous vs Inviscid
- **Outputs:**
  - Wing angle of attack, lift and drag coefficients (alpha, C_L, C_D)
  - Load distribution on wing (lift, drag and moment at a given number of spanwise locations)

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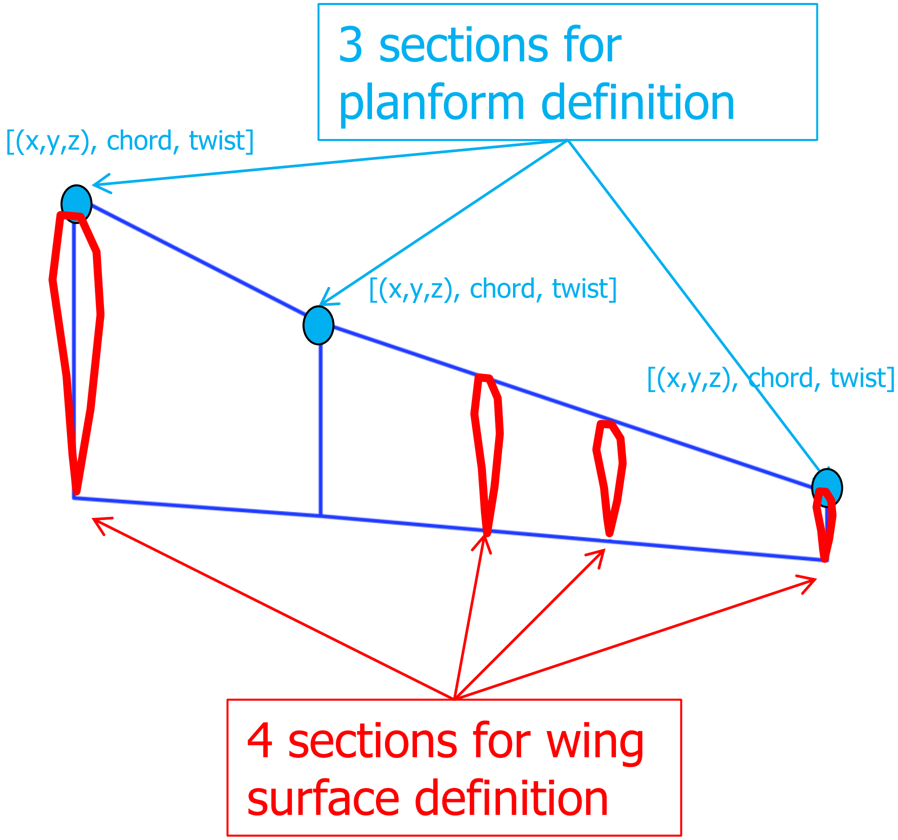
Definition of wing geometry
(Q3D AeroSolver input file)

- Number of sections to define **planform geometry** [(x,y,z), chord, twist]:
  - Min 2 (root and tip)
  - Max: no limits

- Number of the sections to define the **wing surface** (CST coeff):
  - Min 2 (root and tip)
  - Max: no limits



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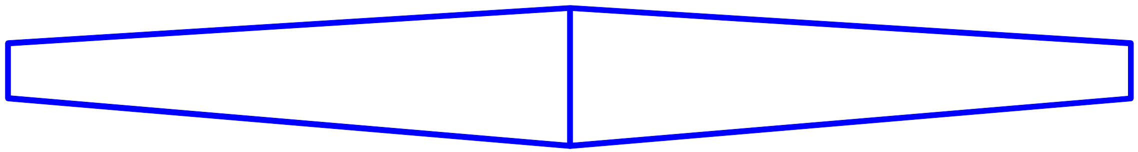
Viscous vs Inviscid Analysis

- Inviscid Analysis
  - Only VLM is used
  - Output includes lift, induced drag and pitching moment distribution
  - Fast but inaccurate
  - Only useful for estimation of the loads needed for weight estimation
- Viscous Analysis
  - Both VLM and viscous airfoil analysis are used
  - Output includes lift, pitching moment and all the drag components

Test Cases for Q3D AeroSolver (low speed)

- **Exercise 1**:
  - Low speed wing: M = 0,2   Re = 1,14e7, H=0m, V=68m/s
  - E553 airfoils for root and tip



Plot spanwise distribution of lift and drag for Alpha = 2 deg

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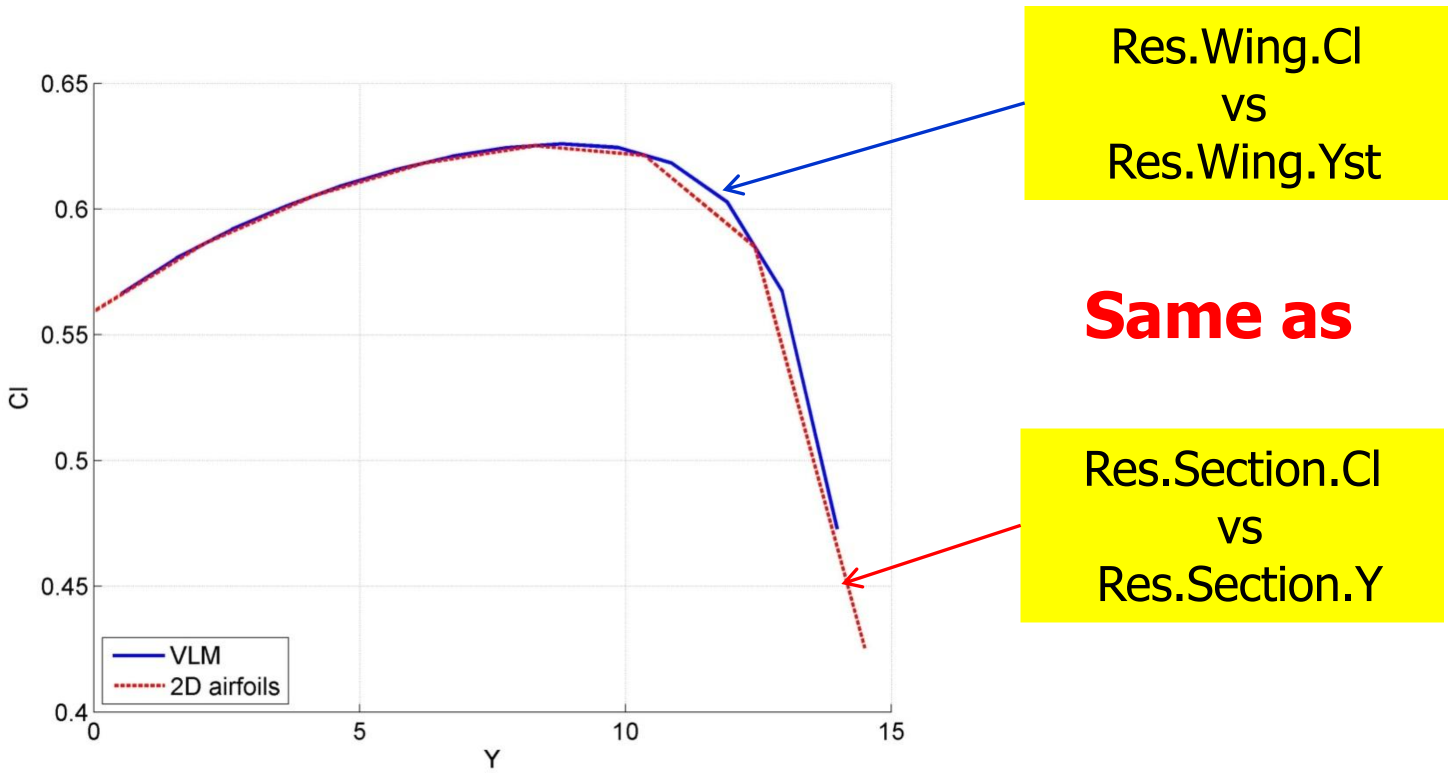
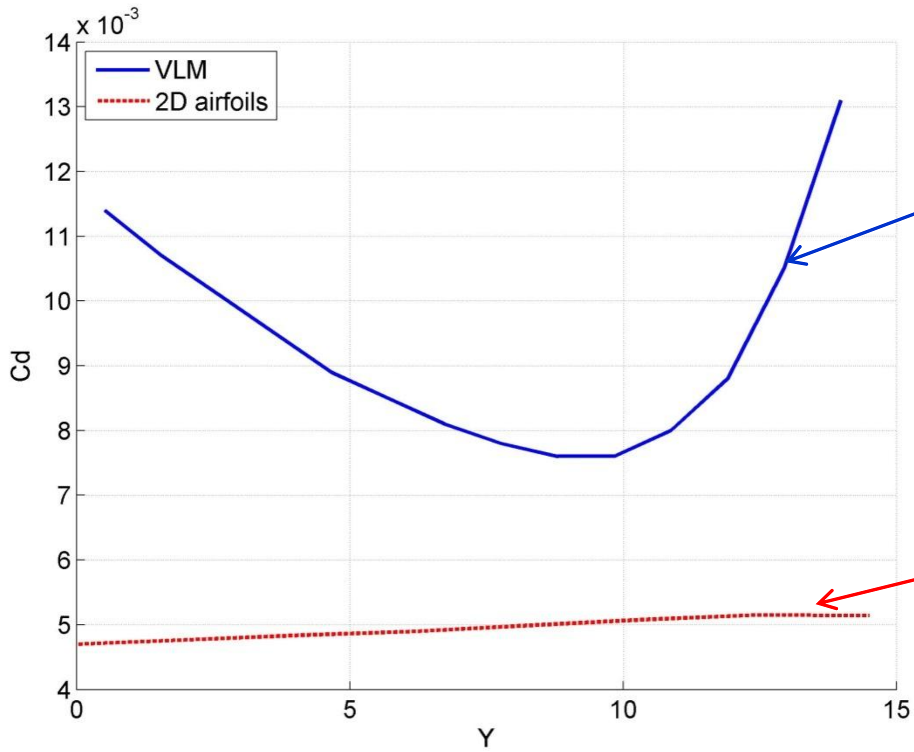
Lift Distribution



The wing lift distribution calculated by VLM is the same as the lift distribution calculated by airfoil analysis

Total Lift: Res.CLwing = 0.5967

Drag Distribution



Res.Wing.cdi
vs
Res.Wing.Yst

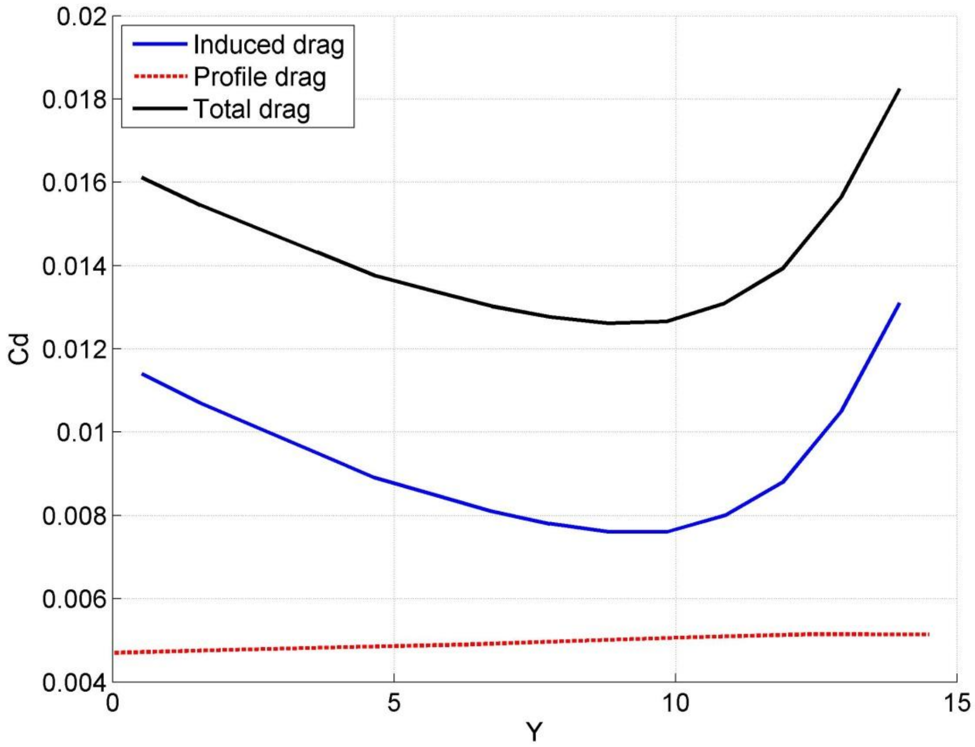
Different from

Res.Section.Cd
vs
Res.Section.Y

Res.Wing.cdi is the wing induced drag

Res.Section.Cd is the airfoils profile drag

Drag Distribution



Total Drag: Res.CDwing = 0.0143

Test Cases 2: Creating Drag Polar

- **Exercise 2:**

Create the drag polar for low speed wing:

1- Calculate CD for CL between 0 and 1

2- Fit a 2nd order polynomial to CL-CD to generate the well know drag polar formula: CD = CDmin + K*(CL-CLmin)^2

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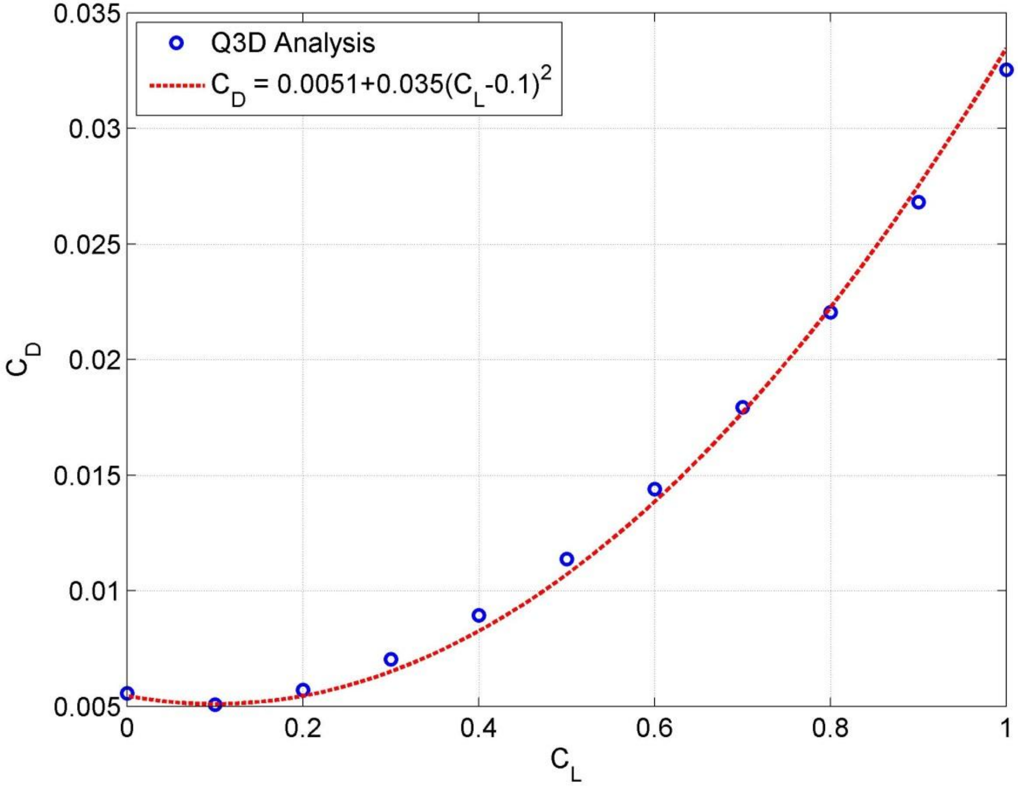
Drag Polar



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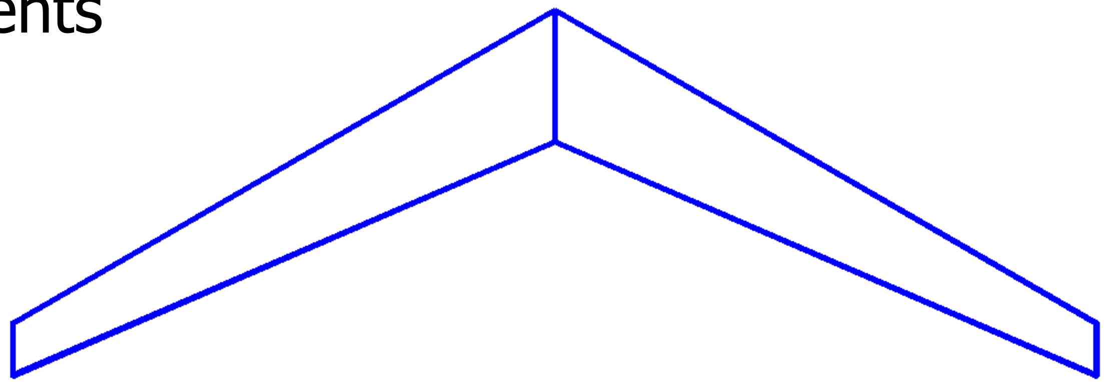
Test Cases for Q3D AeroSolver (high speed)

- **Exercise 3**
- Add 30 deg leading edge sweep to previously defined low speed wing
- Use Withcomb-135, first with 5 and then 10 CST coefficients



Generate geometry of high speed wing

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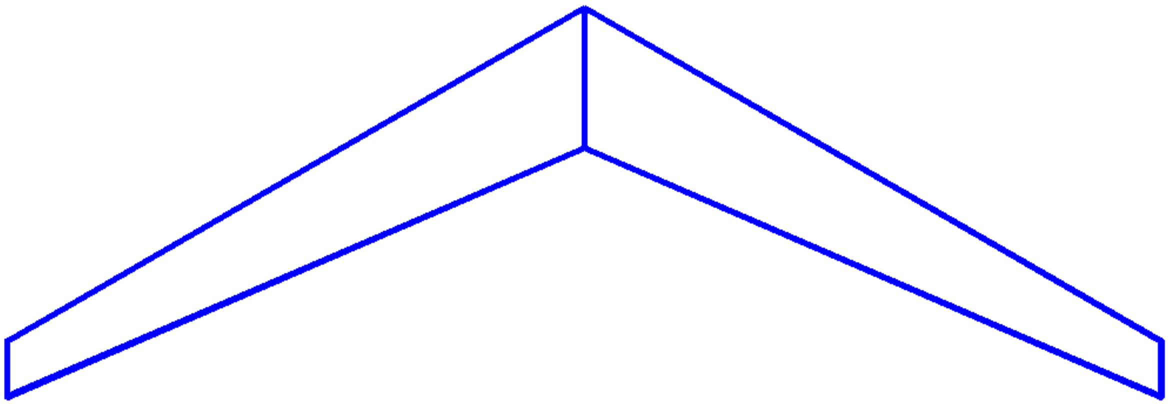
Test Cases for Q3D AeroSolver (high speed)

- **Exercise 3 (continued)**

Compute alpha and CD for the following conditions:

M = 0,8, Re = 1,7e7, CL = 0,4, H = 11000m, V = 236m/s



Exercise 3: Results

- Results for airfoils with 5 CST coefficients:
  - CL = 0,4
  - Alpha = -2,4
  - CD = 0,0131

Why Negative AoA?

- Results for airfoils with 10 CST coefficients:
  - CL = 0,4
  - Alpha = -2,26
  - CD = 0,0124

Why different values?

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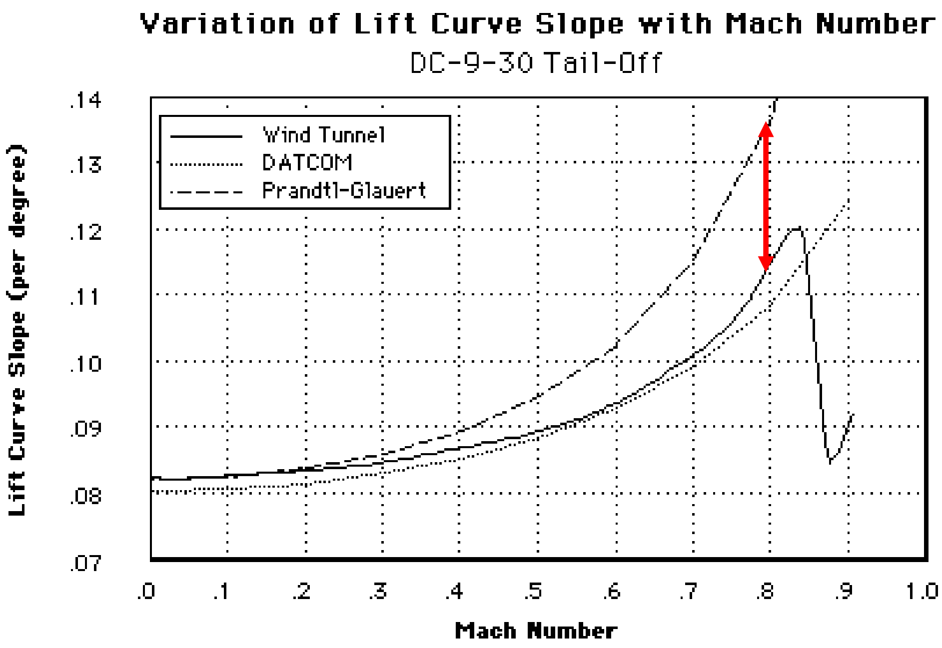
Why negative AOA?

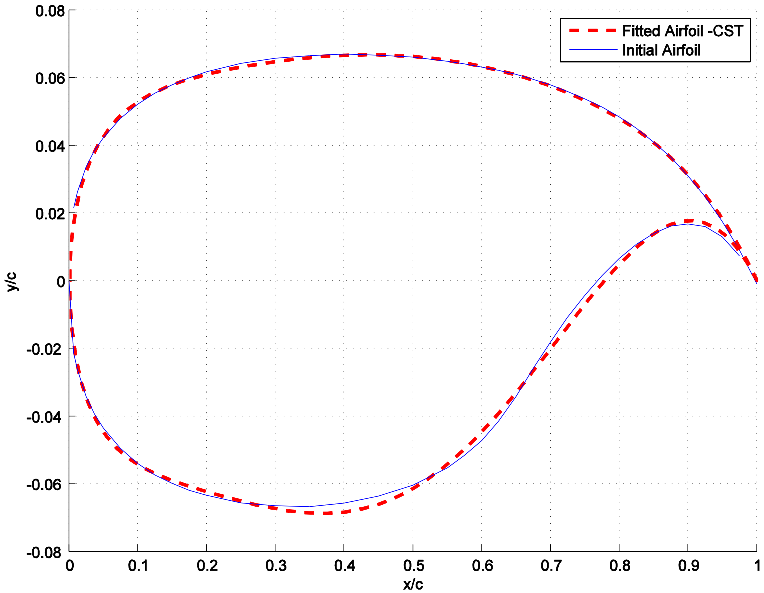
- VLM uses Prandtl-Glauert compressibility correction:
  - Prandtle-Glauert overestimates lift at high Mach numbers
  - As a result: for a given lift the AOA is underestimated



Why different Results?

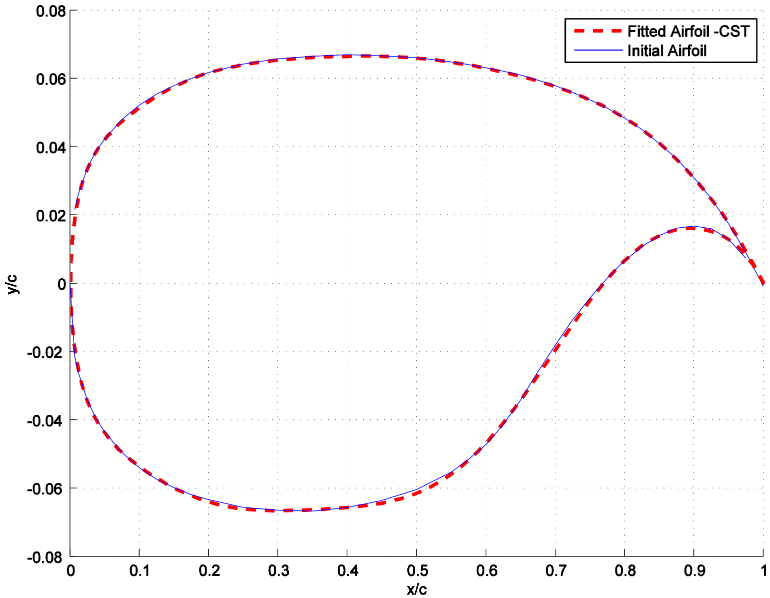
5 CST Coef.

Eu = 1,9e-5 , El = 7,7e-5



10 CST Coef.

Eu = 2,5e-6, El = 9,7e-6



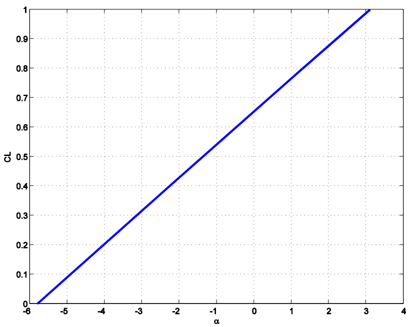
Eu = Error of curve fitting to the upper surface

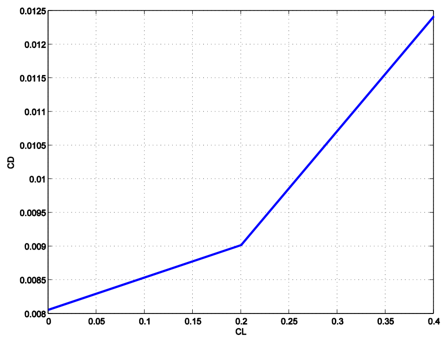
El = Error of curve fitting to the lower surface

Generation of Drag Polar for the Transonic Wing

Exercise 4:

Create drag polar for the wing used in exercise 3
(Calculate Alpha and CD for CL between 0 and 1)





CL =	[0	0,2	0,4	0,6	0,8	1]
CD =	[0.0081	0.0090	0.0124	NaN	NaN	NaN]

Whay NaN?

Effect of Airfoil Shape

Exercise 5:

Perform the same analysis as in Exe 3, using the Eppler airfoil (E553) in place of the Withcomb.

- CL = 0,4
- Alpha = -1,5
- CD = NaN

Why NaN?

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Why NaN?

VGK is based on the full-potential equation.

Assumption behind the full potential equation:

Flow is irrotational

Possible sources for rotation:

1- Viscousity

2- Shock wave

In case of strong shock waves (Mach number in front of the shock higher than 1.25-1.3) the solution of the full potential equation is not fully trustable.

In case of using VGK:

In high Mach numbers: high values of airfoil Cl (exercise 4) and/or not suitable airfoil shape (exercise 5) can make difficulties for VGK to converge

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