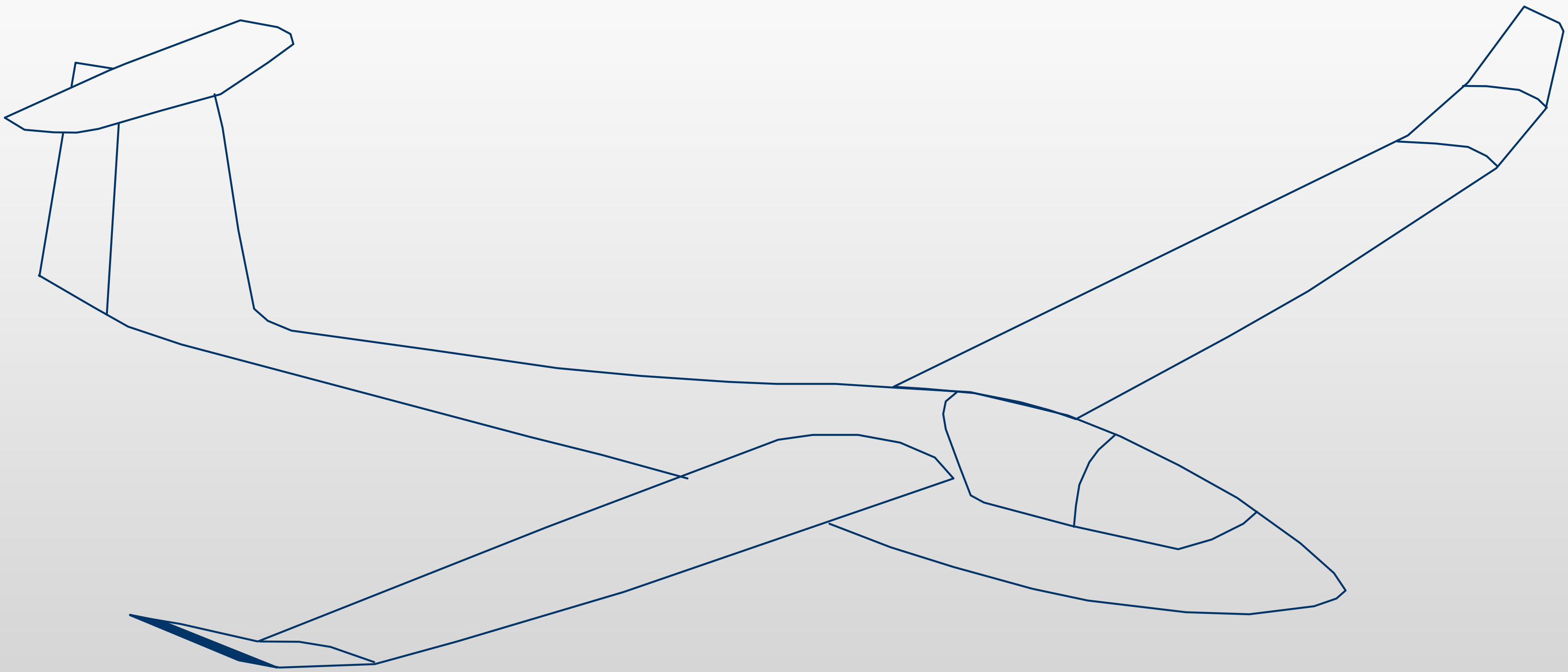
Illustration of the use of Control Polars in XFLR5



XFLR5 v4.07 Beta

Control Polars

- XFLR5 v4.07 introduces the Type 5 & Type 6 polars for plane analysis
- The idea is to plot performance data for balanced flight conditions, as a function of control settings
- By controls, we mean
  - CG position
  - Wing incidence angle
  - Elevator incidence angle
  - Flap angles
- For each set of control data, the code finds by iteration the angle of attack (aoa) such that the total pitching moment GCm=0, and stores the corresponding OpPoint

Control analysis

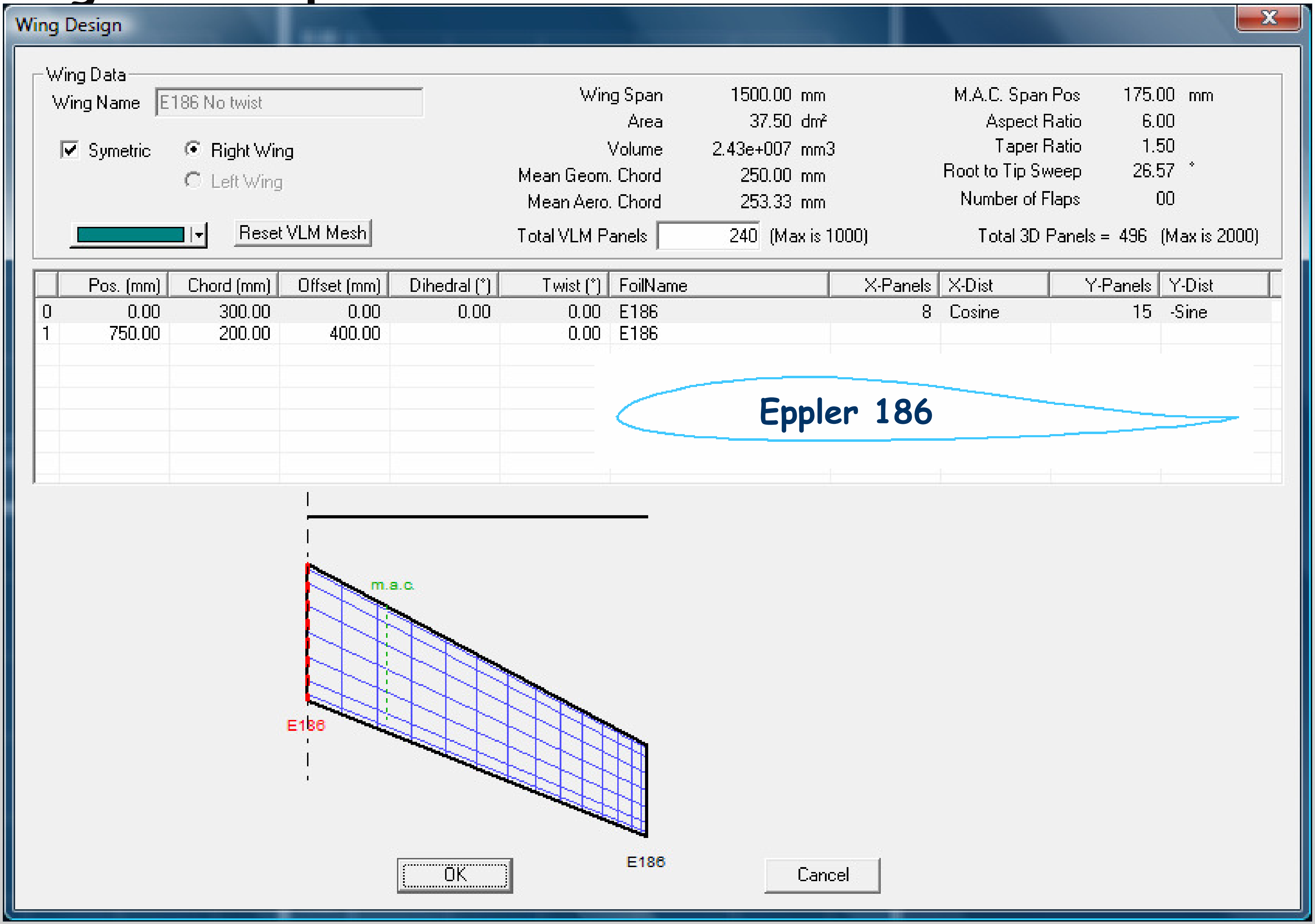
- This presentation illustrates the use of control polars for three test cases :

  Case 1: Analysis of the performances of a flying wing vs. CG position

  Case 2a: Analysis of the performance of a plane as a function of elevator incidence

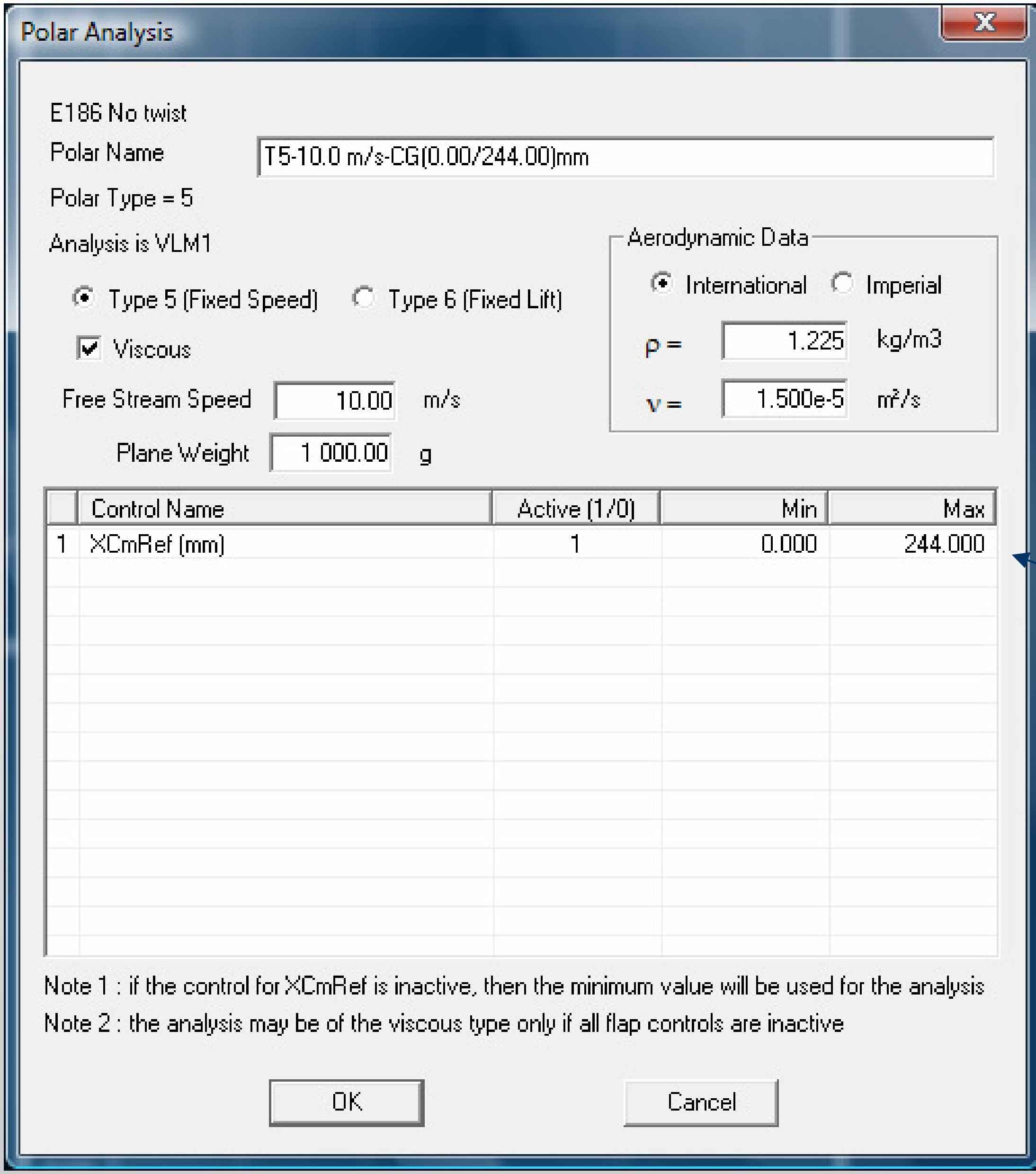
  Case 2b: Analysis of the speed of a plane as a function of wing flap settings

1st case : Analysis of the performances of a flying wing vs. CG position



	Pos. (mm)	Chord (mm)	Offset (mm)	Dihedral (°)	Twist (°)	FoilName	X-Panels	X-Dist	Y-Panels	Y-Dist
0	0.00	300.00	0.00	0.00	0.00	E186	8	Cosine	15	-Sine
1	750.00	200.00	400.00		0.00	E186				

1st case : Control polar data for study of CG position

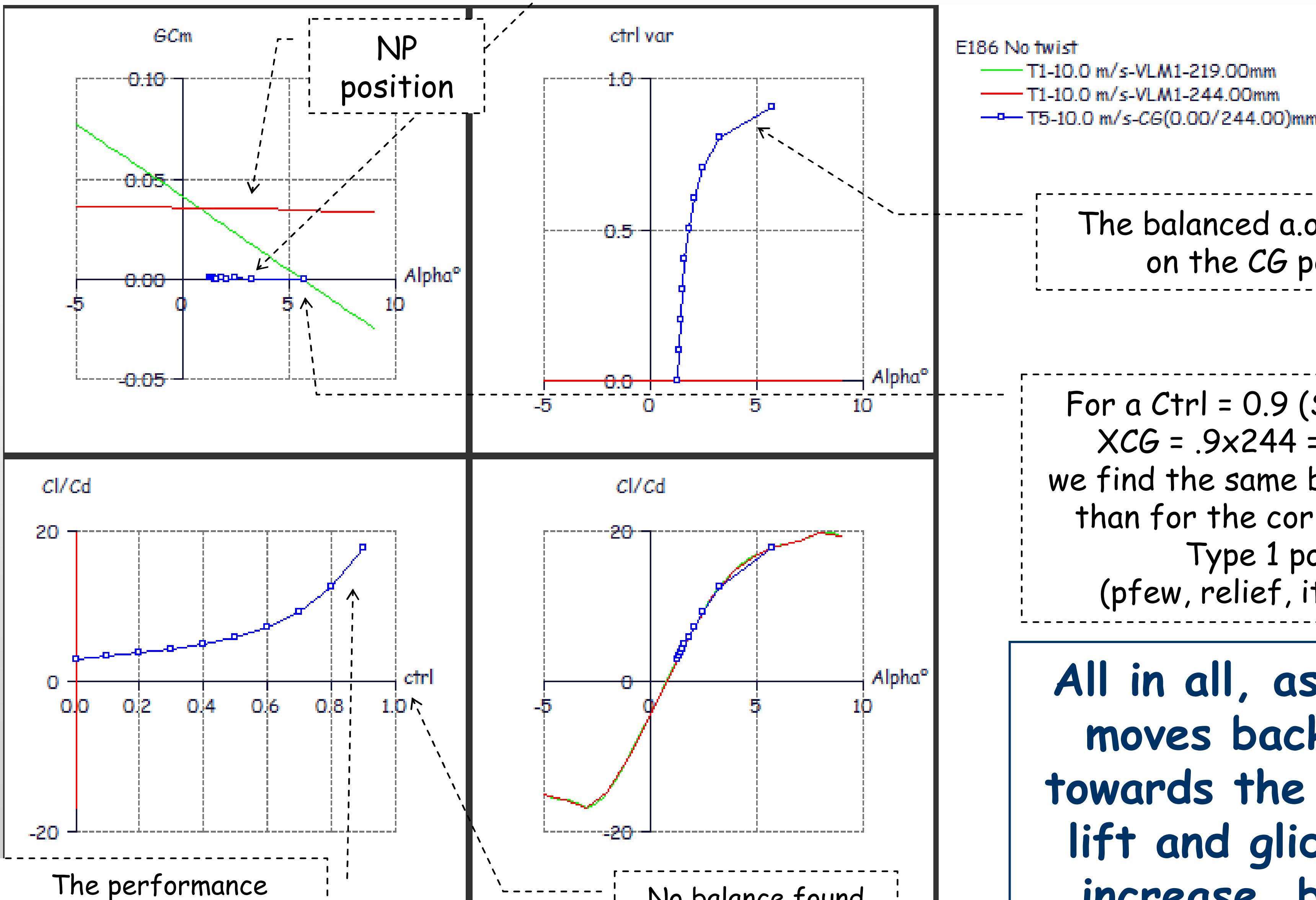


The only active control is the CG's position which varies from 0 up to the Neutral Point (NP) position

Results in the polar's 4-graph view

Alpha has been calculated to achieve zero pitching moment

NP position



The balanced a.o.a. depends on the CG position

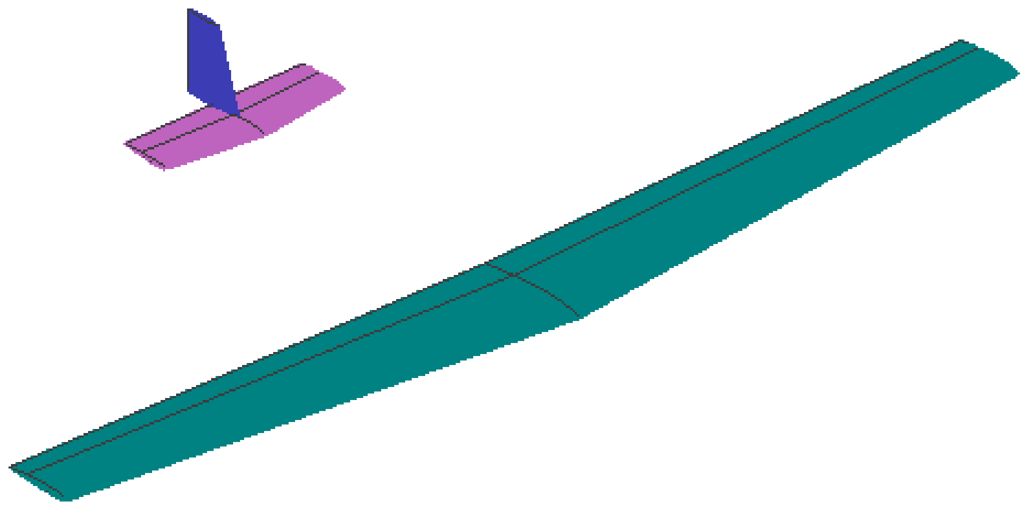
For a Ctrl = 0.9 (SM10%), i.e XCG = .9x244 = 219 mm, we find the same balance point than for the corresponding Type 1 polar (pfew, relief, it works !)

All in all, as the CG moves backwards towards the NP, the lift and glide ratio increase, but the stability and speed decrease

The performance increases as the CG is moved backwards

No balance found for Ctrl = 1.0, i.e $X_{CG} = X_{NP}$ ... Makes sense

2nd case : Control Polar as a function of elevator incidence



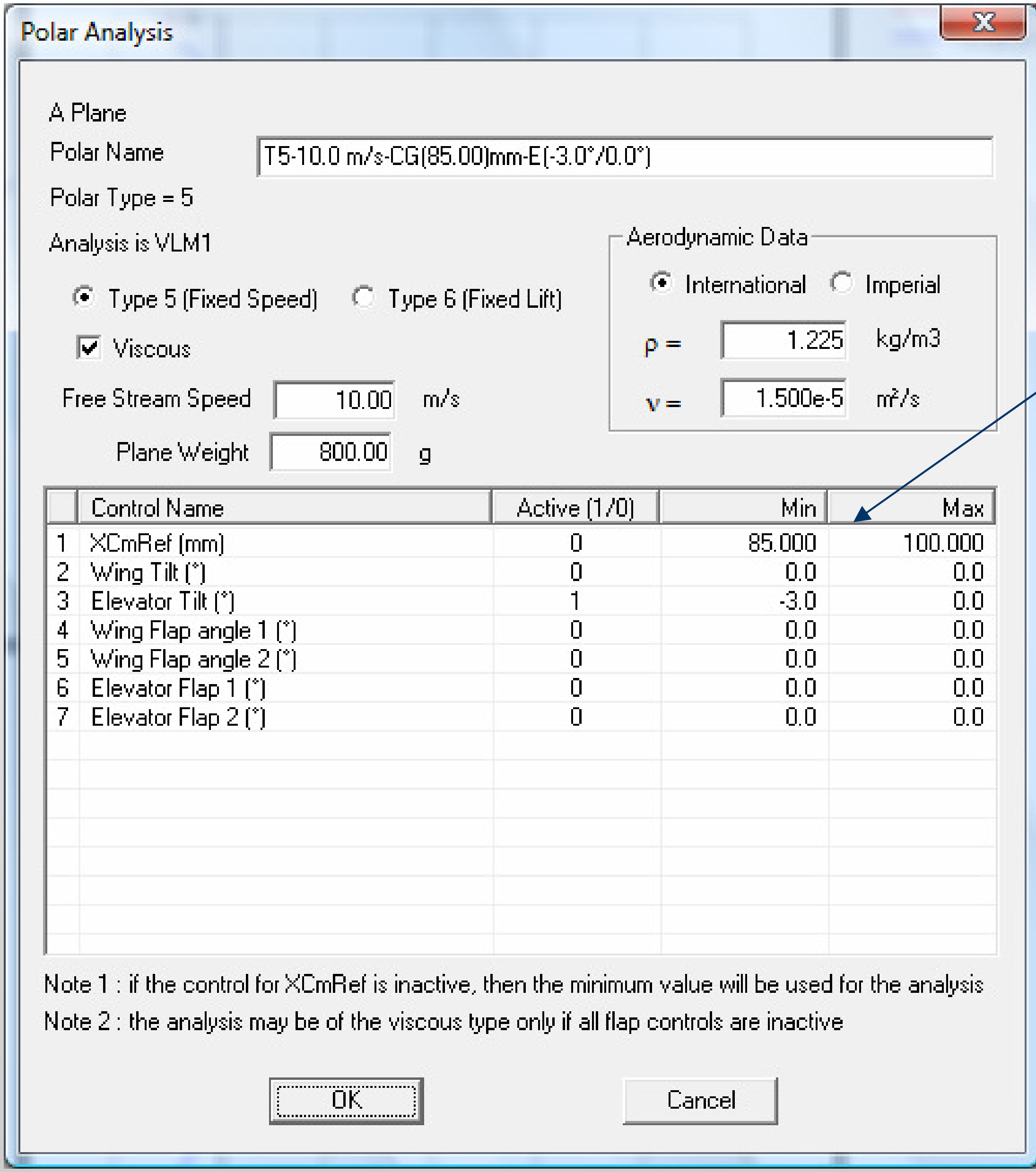
A Plane

Wing span	=	1600.00 mm
Wing area	=	23.20 dm²
Plane weight	=	1000.00 g
Wing load	=	43.103 g/dm²
Tail Volume	=	0.46
Root chord	=	180.00 mm
M.A.C.	=	147.82 mm
Twist at tip	=	0.0 °
Aspect Ratio	=	11.0
Taper Ratio	=	1.6
Rt-Tip sweep	=	3.8 °

This time, we want

- First to check performances vs. elevator incidence
- Next to study the speed vs. the wing's flaps position

Case 2a : Control Polar as a function of elevator incidence



Polar Analysis

A Plane

Polar Name: T5-10.0 m/s-CG(85.00)mm-E(-3.0°/0.0°)

Polar Type = 5

Analysis is VLM1

(•) Type 5 (Fixed Speed) ( ) Type 6 (Fixed Lift)

[x] Viscous

Free Stream Speed 10.00 m/s

Plane Weight 800.00 g

Aerodynamic Data

(•) International ( ) Imperial

ρ = 1.225 kg/m3

ν = 1.500e-5 m²/s

	Control Name	Active (1/0)	Min	Max
1	XCmRef (mm)	0	85.000	100.000
2	Wing Tilt (°)	0	0.0	0.0
3	Elevator Tilt (°)	1	-3.0	0.0
4	Wing Flap angle 1 (°)	0	0.0	0.0
5	Wing Flap angle 2 (°)	0	0.0	0.0
6	Elevator Flap 1 (°)	0	0.0	0.0
7	Elevator Flap 2 (°)	0	0.0	0.0

Note 1 : if the control for XCmRef is inactive, then the minimum value will be used for the analysis

Note 2 : the analysis may be of the viscous type only if all flap controls are inactive

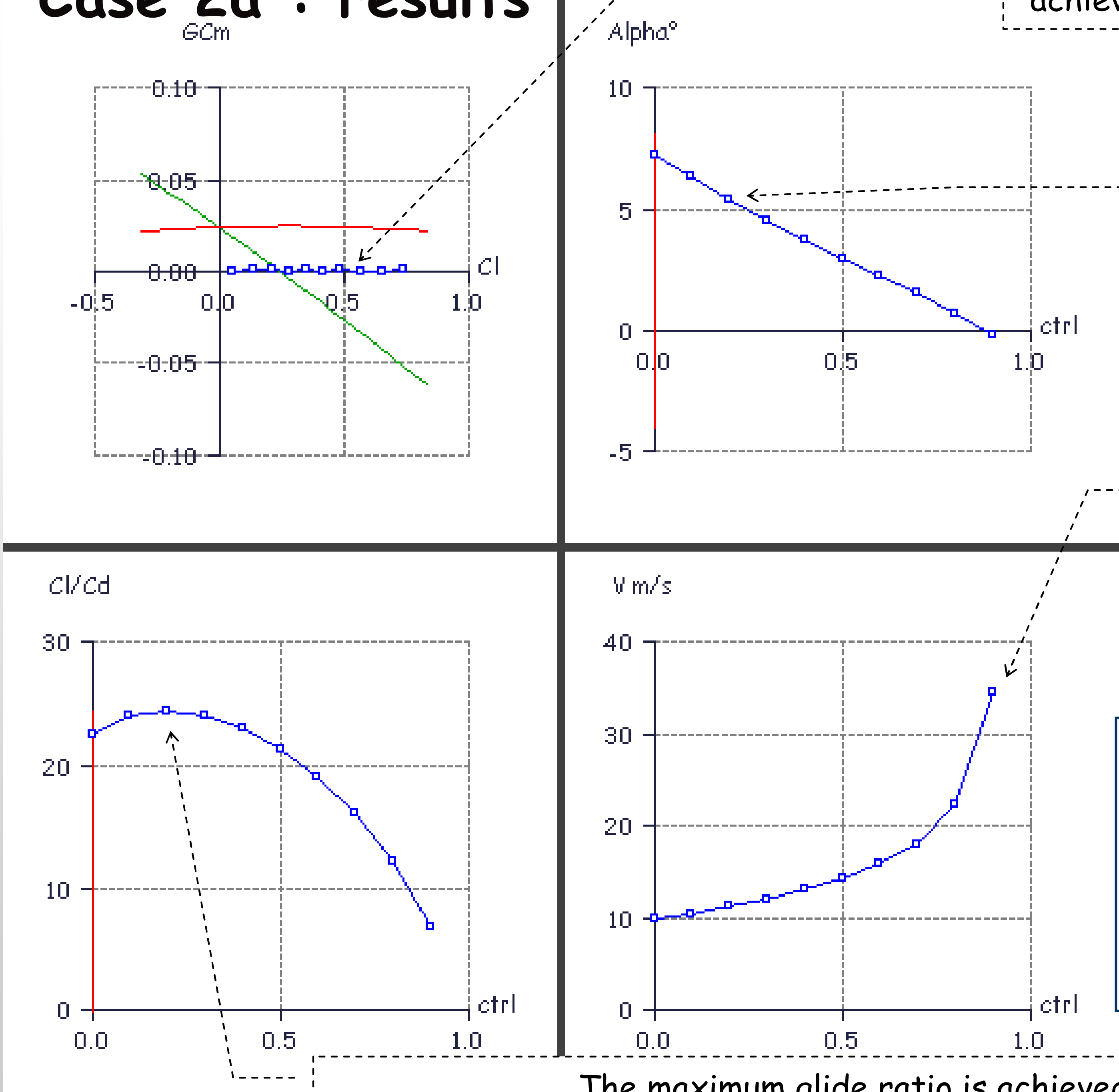
OK Cancel

The CG position is inactive, therefore the program will use the min value of 85 mm, which corresponds to a Static Margin = 10%

The only active control is the elevator incidence, which varies from -3° to 0°

Case 2a : results

Alpha has been calculated to achieve zero pitching moment



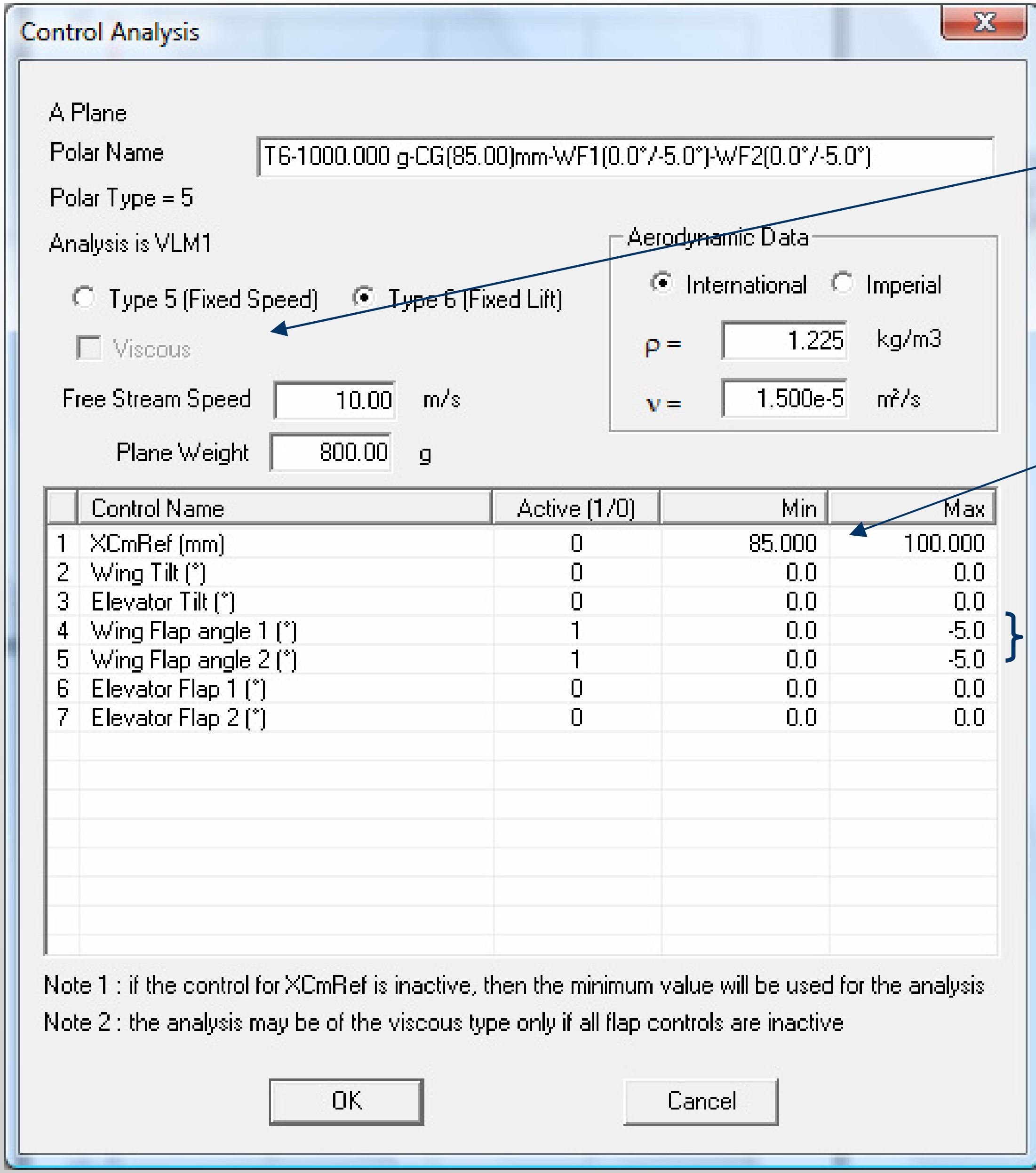
For ctrl=0.2, the angle of attack is roughly 5.5° : Will fly slowly, not pleasant

The speed increases as the elevator's incidence tends towards 0°

Need to choose between glide capacity and speed

The maximum glide ratio is achieved for ctrl=0.2, i.e : Elevator Tilt = -3 x (1-0.2) = -2.4°

2b : Maximize speed as a function of flap settings



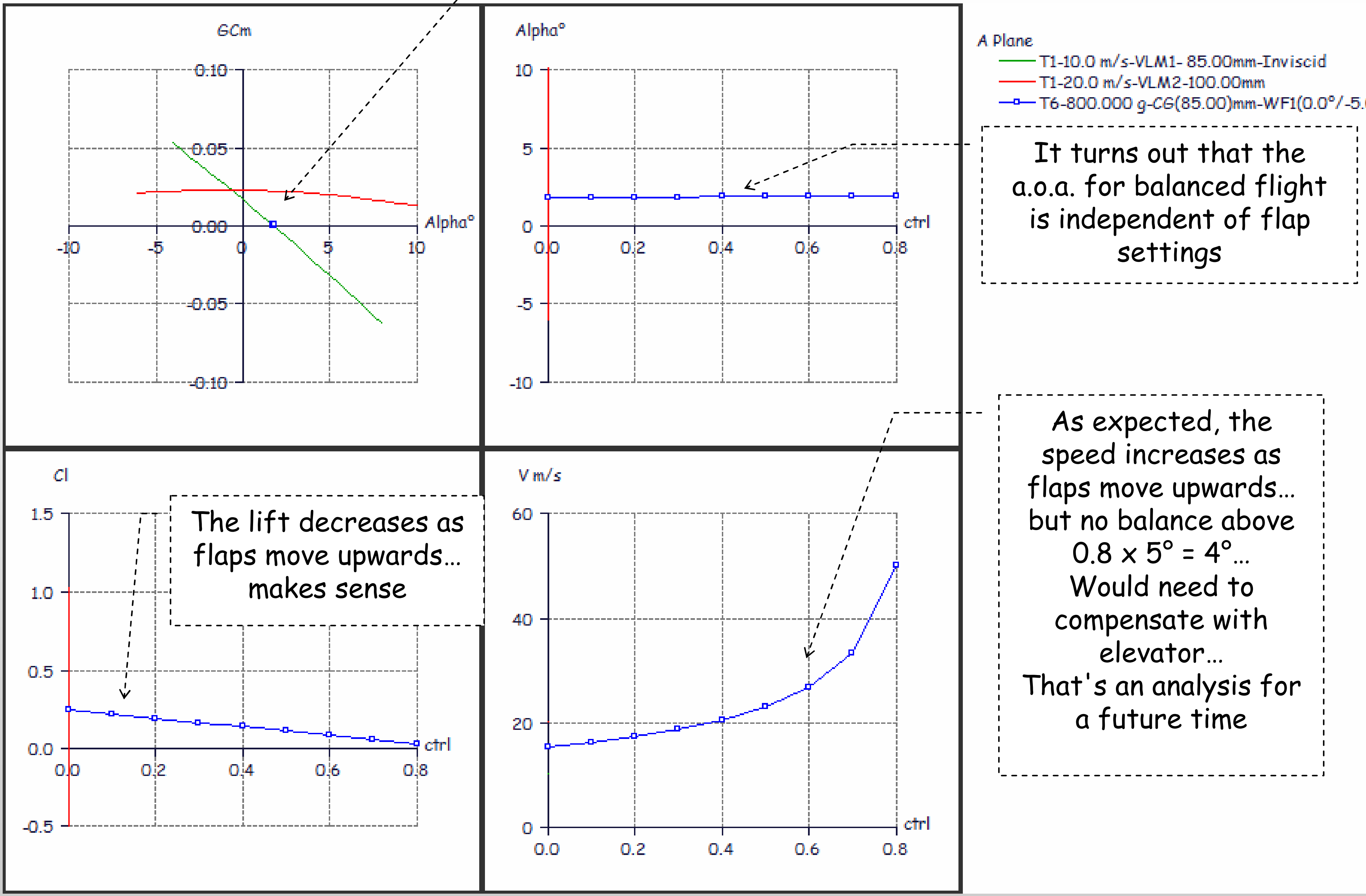
We don't have foil polar meshes available for all flap positions, so the analysis is necessarily non-viscous

The CG position is inactive and fixed at 85 mm, which corresponds to a Static Margin = 10%

The flap's angle varies from 0° to 5° upwards – same for both wings

Case 2b : Results

Alpha has been calculated to achieve zero pitching moment



It turns out that the a.o.a. for balanced flight is independent of flap settings

The lift decreases as flaps move upwards… makes sense

As expected, the speed increases as flaps move upwards… but no balance above 0.8 x 5° = 4°… Would need to compensate with elevator… That's an analysis for a future time