Airbus A380 *The SCUFF*

Chase Ashton Doug Hillson Dave Simon

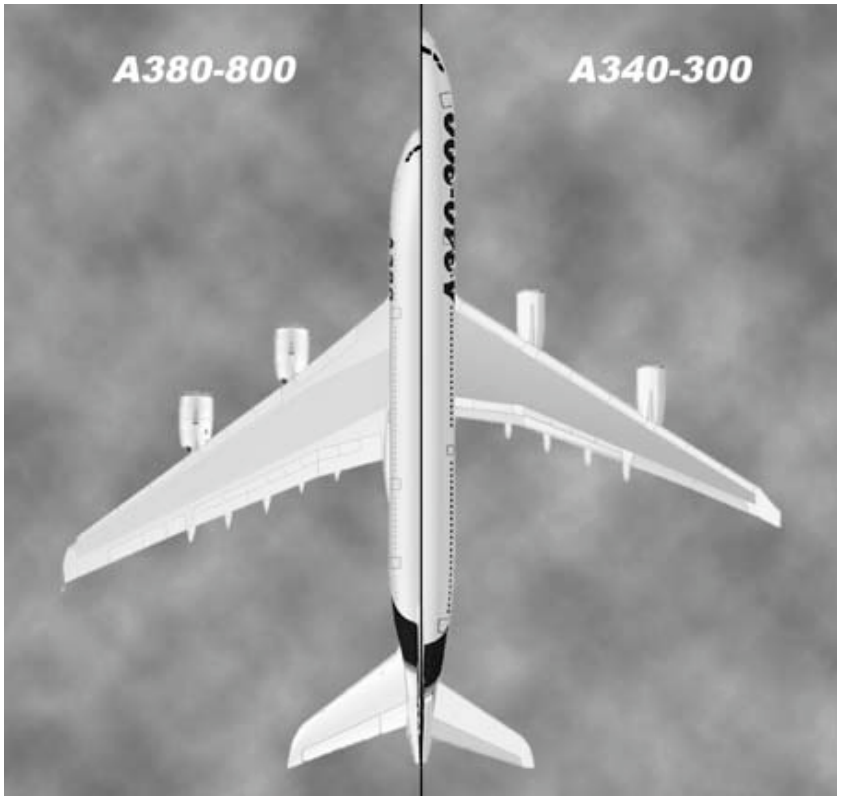
Gate Box Requirement

A380 AR = 7.53
A340 AR = 9.21

$\left.\frac{L}{D}\right)_{\max}$ is proportional to $\sqrt{AR}$ and
R is proportional to $\sqrt{AR}$

$$\frac{\sqrt{9.21}}{\sqrt{7.53}} = 1.106$$

Range and L/D would be **%10.6 greater**

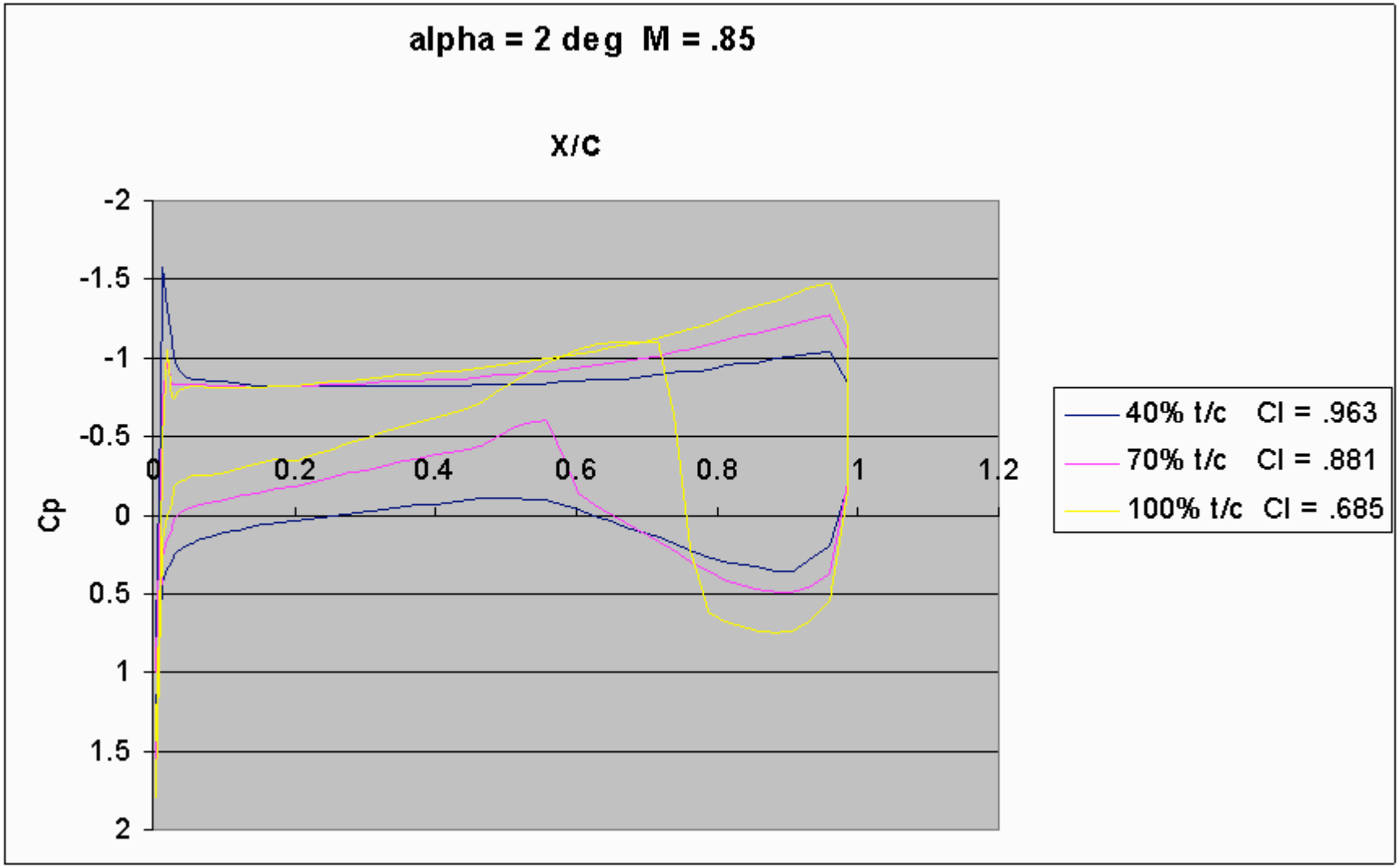


If the A380 and A340 had the same wingspan

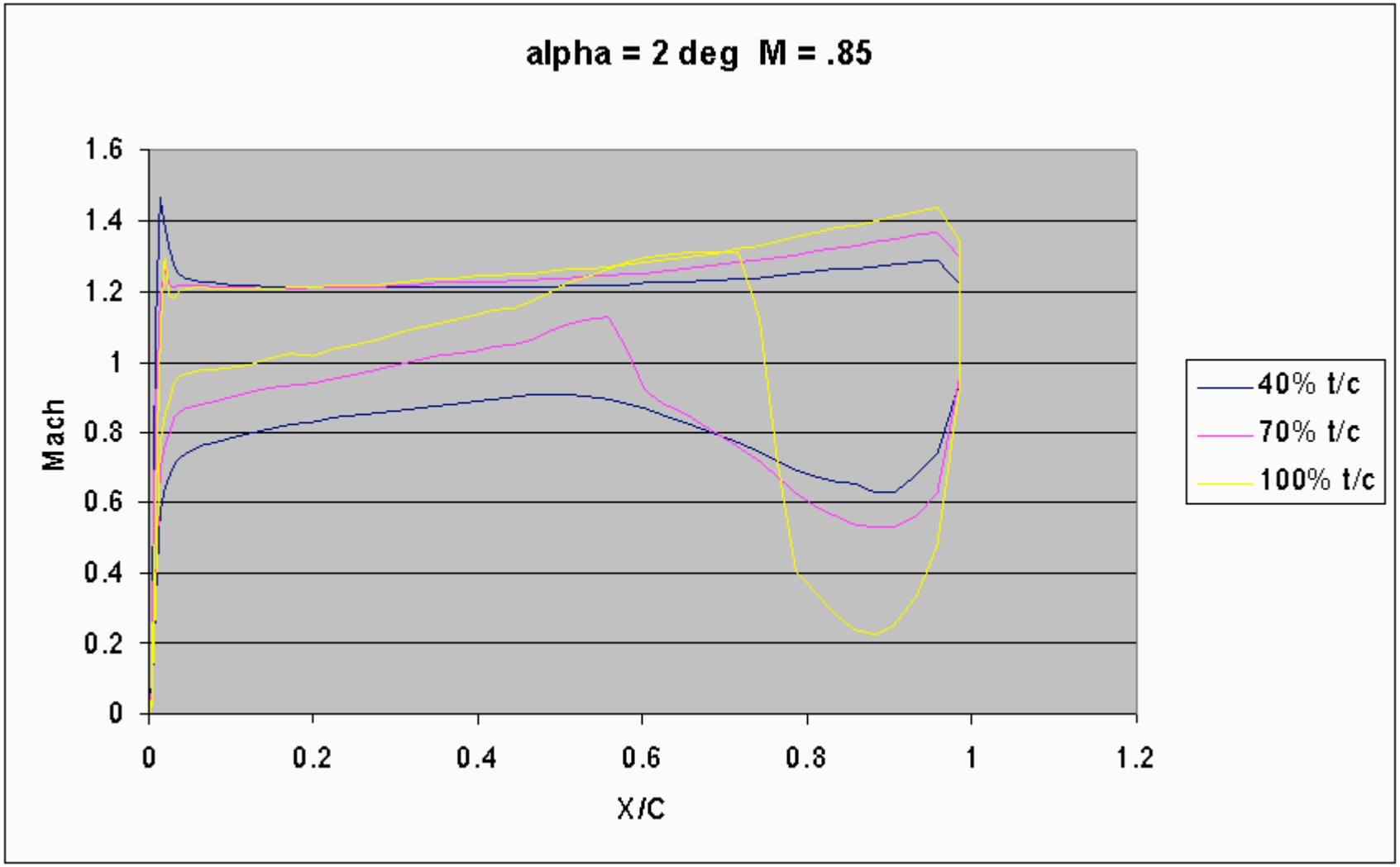
Airfoil Selection

- Supercritical airfoils considered
  - Whitcomb and others
- Transonic Helicopter blades considered
  - NLR-7223-43
- Boeing airfoils considered
  - Only access to old airfoils
  - Company is very secretive
- Choose most recent supercritical airfoil
  - SC(2)-0714
- The SC(2)-0714 airfoil appears relatively easy to manufacture
  - Depending on t/c
- Ran on Tsfoil2
  - Normal Mach number greater than 1.3
- Airbus and Boeing have airfoils not accessible for this project
  - Designed for M = .80 or greater
  - Probably work better than SC(2)-0714

NASA SC(2)-0714 AIRFOIL

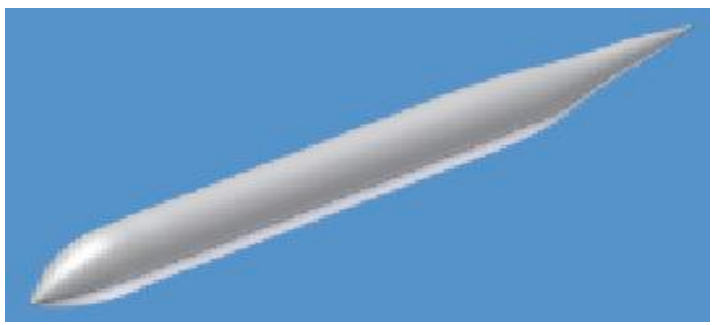


NASA SC(2)-0714 AIRFOIL

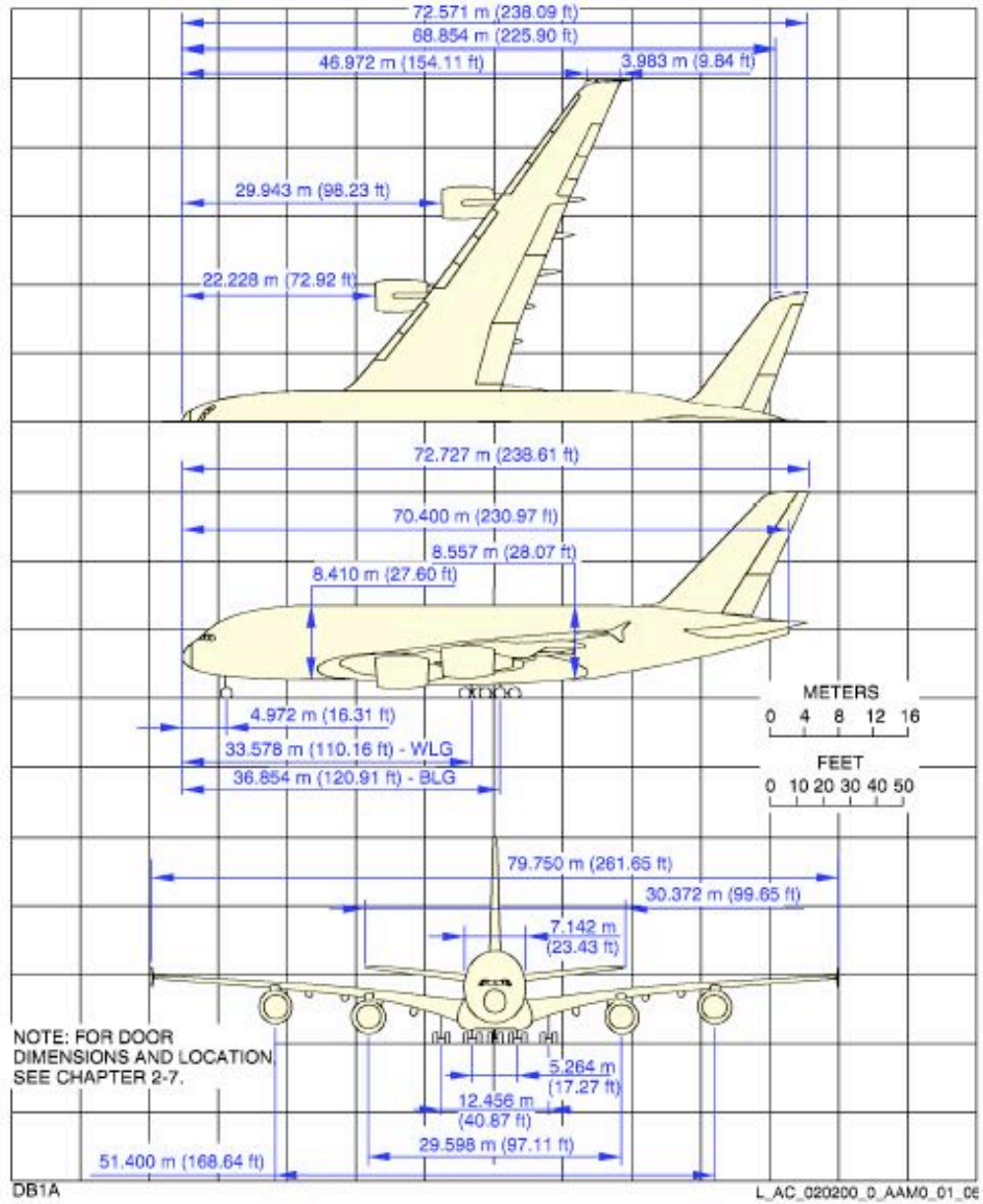


Surface Area Calculation

- Diagram used to model the A380 in CAD
- Complex bodies of revolution modeled in Inventor



- Planar bodies measured in AutoCAD



C_{D_0} Calculation using FRICTION

Input Parameters

Component	Wetted Area (ft^2)	Reference Length (ft)	Thickness/Chord
Fuselage	16050	231	0.119
Wing	18284	41.5	0.082
Horizontal Tail	4211	22.3	0.081
Vertical Tail	2990	29.1	0.093
Engines (4)	3138 (each)	21.6	0.630

Reference Area: 9380 ft^2
All surfaces assumed 100% turbulent flow
Wing, Horizontal, Vertical Tail - Modeled as Planer Surfaces
Fuselage, Engines – Modeled as Bodies of Revolution

C_{D_0} Calculation using FRICTION

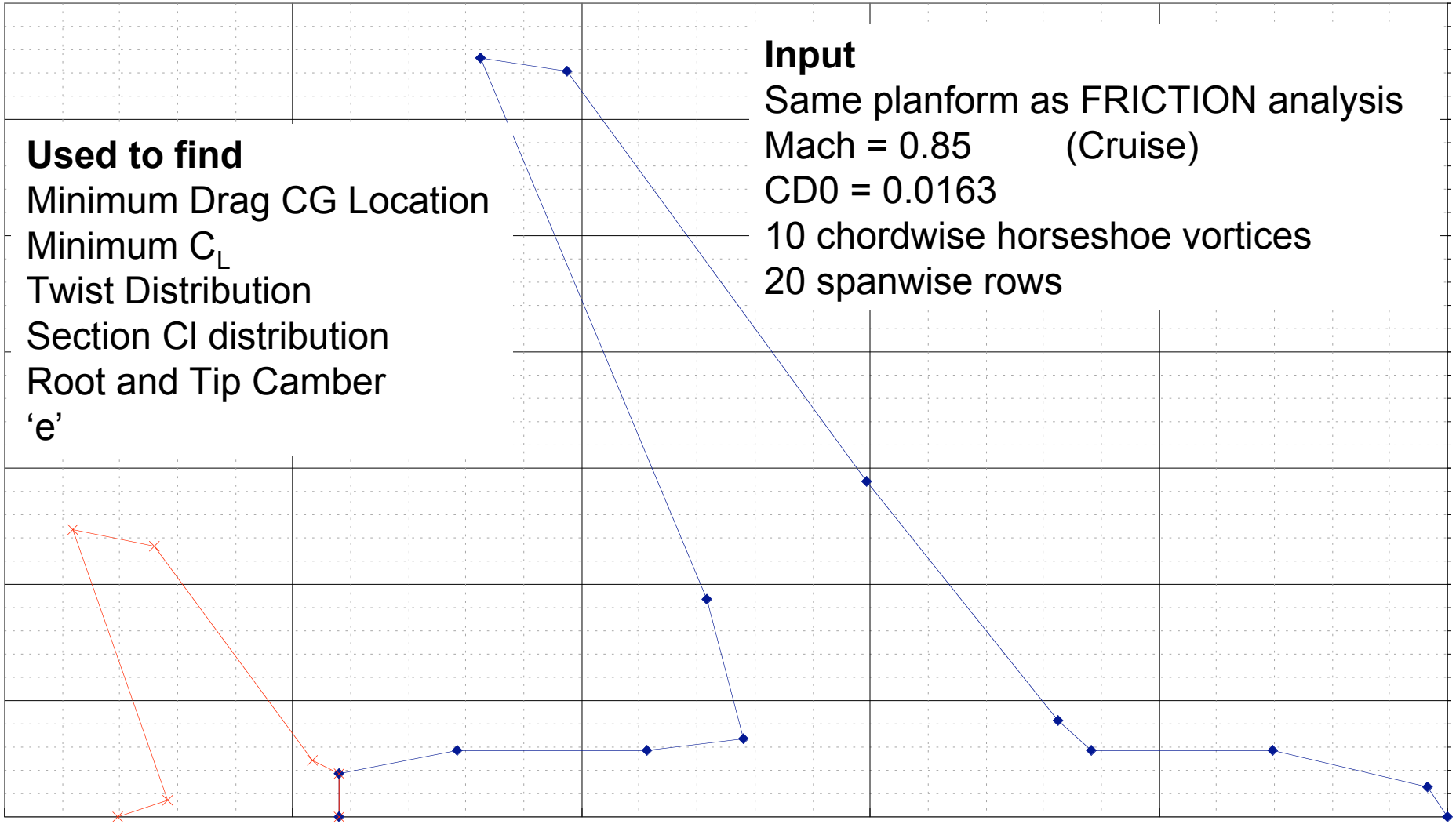
Results

Altitude	Mach Number	CD0	CD_{min}	CL	L/D Max	
5000	0.10	0.0212	0.0424	0.689	16.26	
10000	0.50	0.0167	0.0334	0.611	18.32	
20000	0.75	0.0161	0.0322	0.600	18.68	
25000	0.80	0.0162	0.0324	0.602	18.60	
30000	0.85	0.0163	0.0326	0.605	18.51	Cruise
30000	0.89	0.0162	0.0324	0.601	18.62	
30000	0.90	0.0161	0.0322	0.601	18.65	

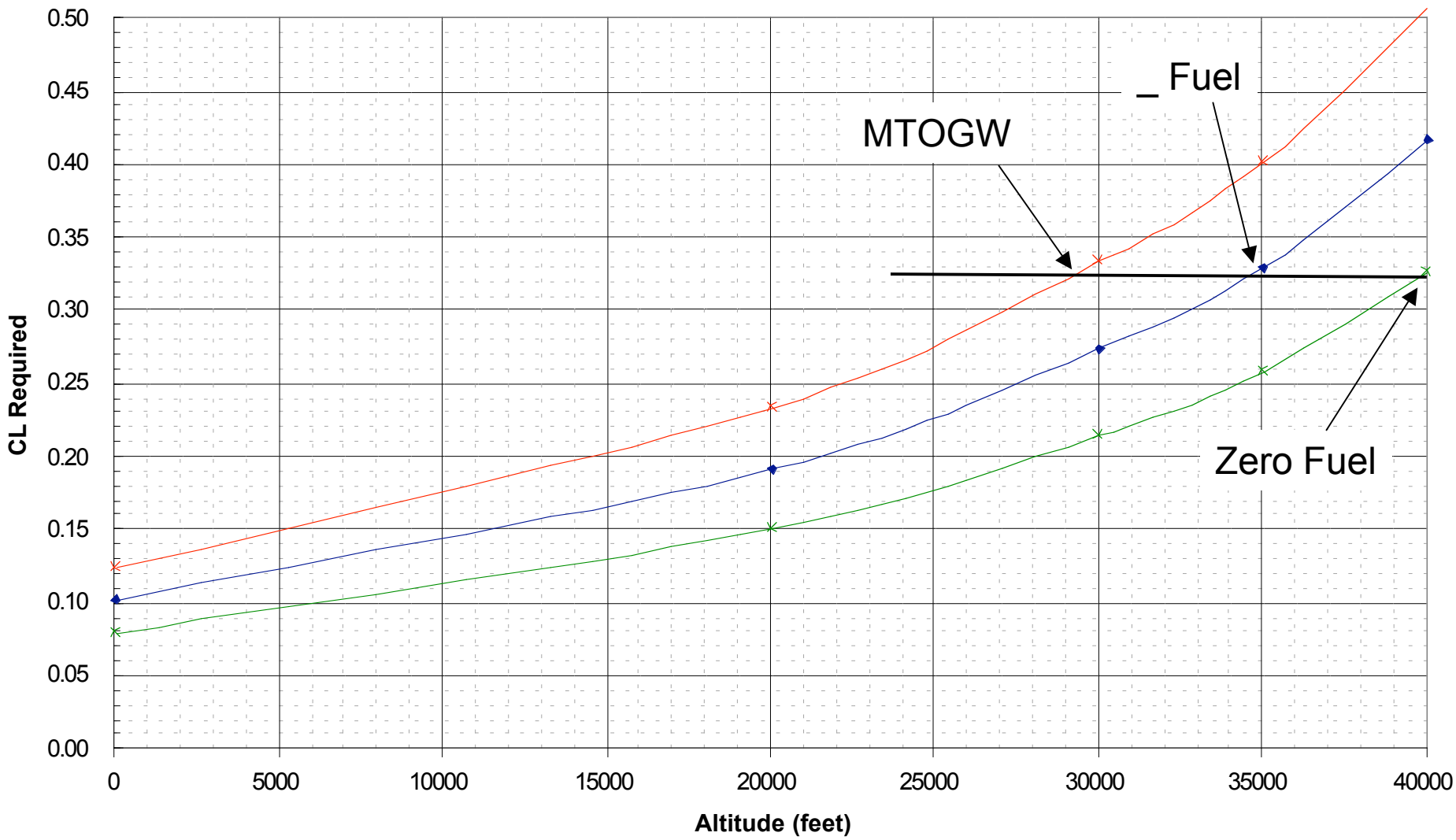
$$C_{D_{MIN}} = 2 \cdot C_{D_0}$$

$$C_L = \sqrt{\pi \cdot AR \cdot e \cdot C_{D_0}}$$

LAMDES

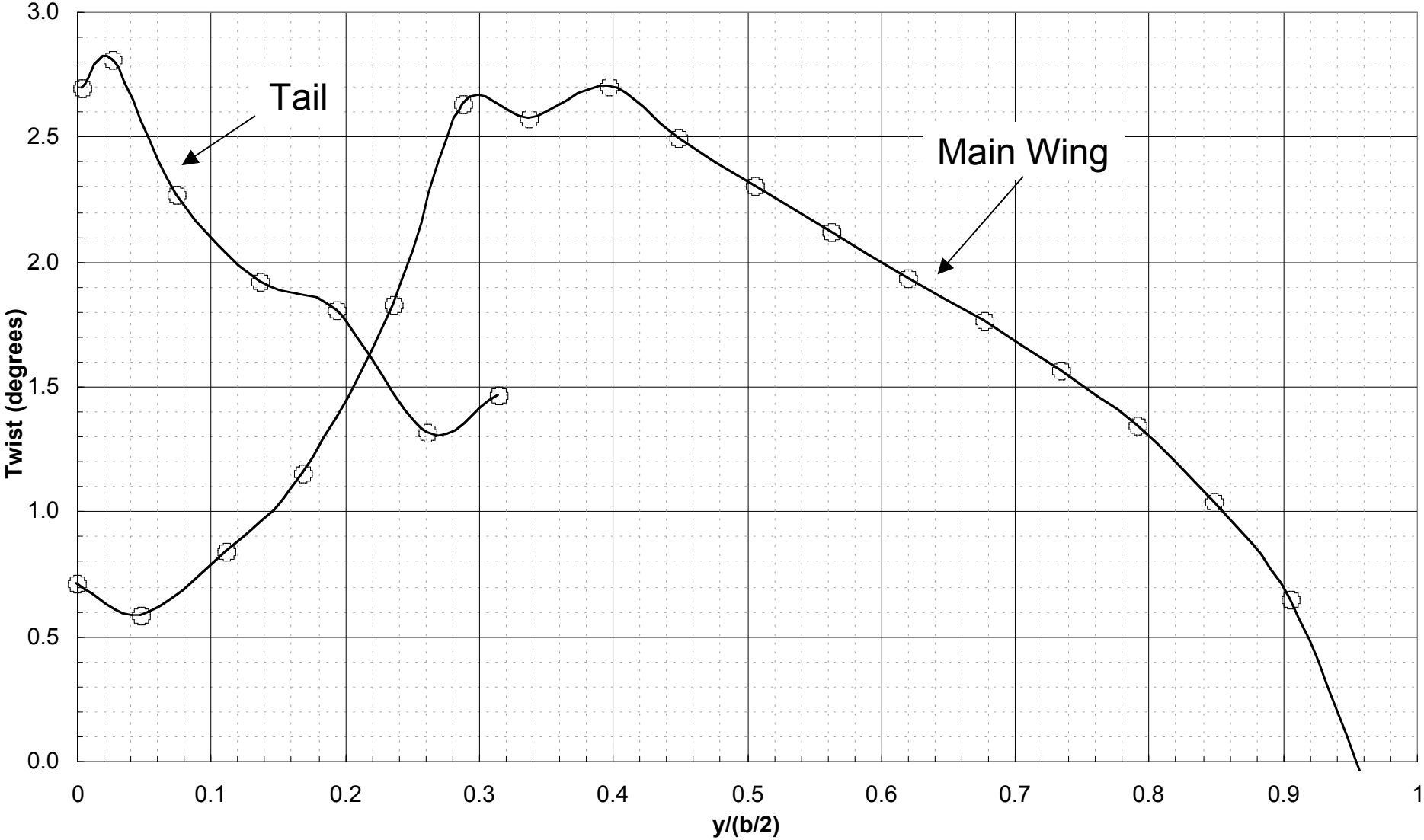

Used to find
Minimum Drag CG Location
Minimum C_L
Twist Distribution
Section Cl distribution
Root and Tip Camber
'e'
Input
Same planform as FRICTION analysis
Mach = 0.85 (Cruise)
CD0 = 0.0163
10 chordwise horseshoe vortices
20 spanwise rows

Best Cruise Altitude at M=0.85

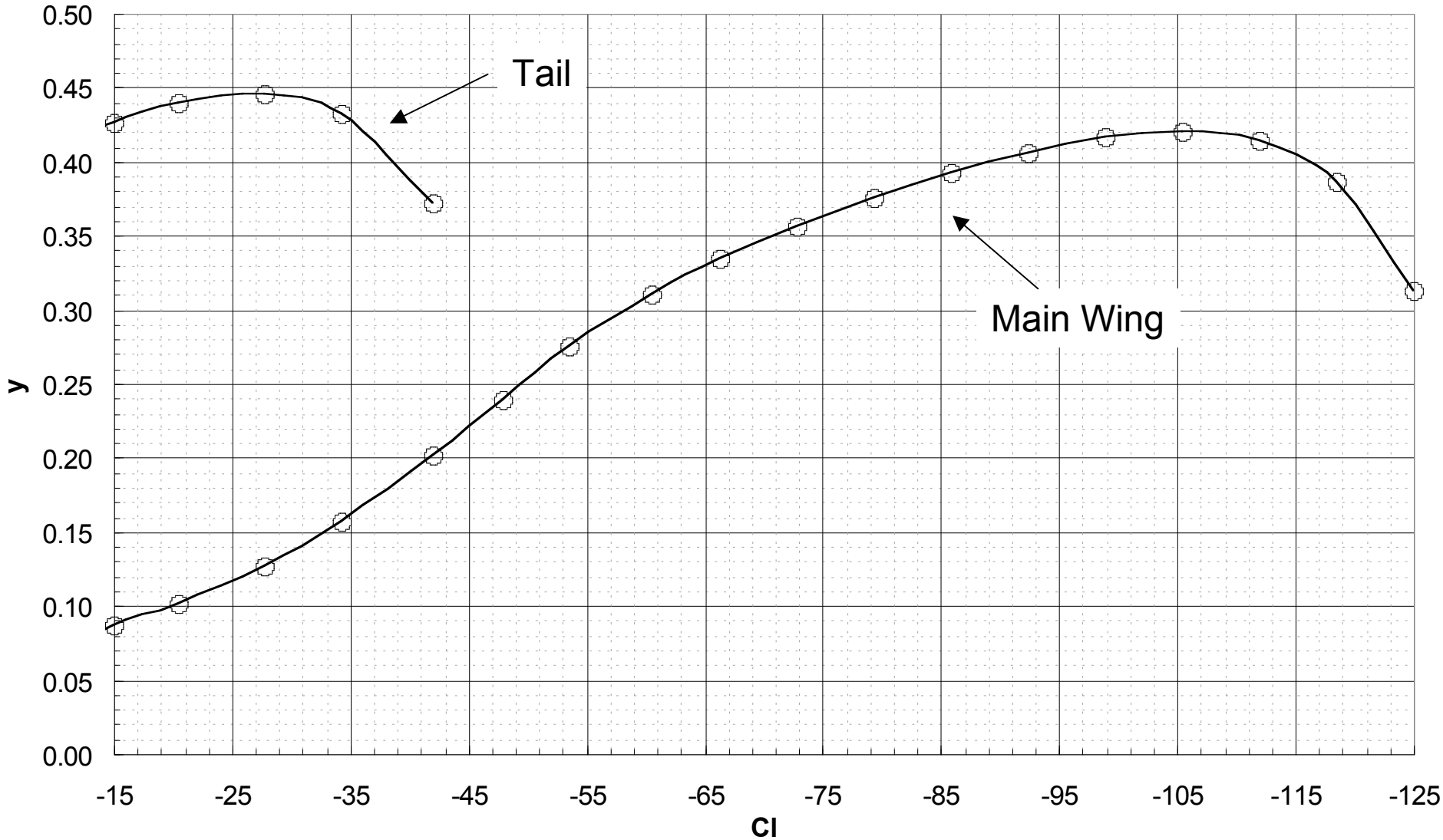


Linear Theory Twist Distribution

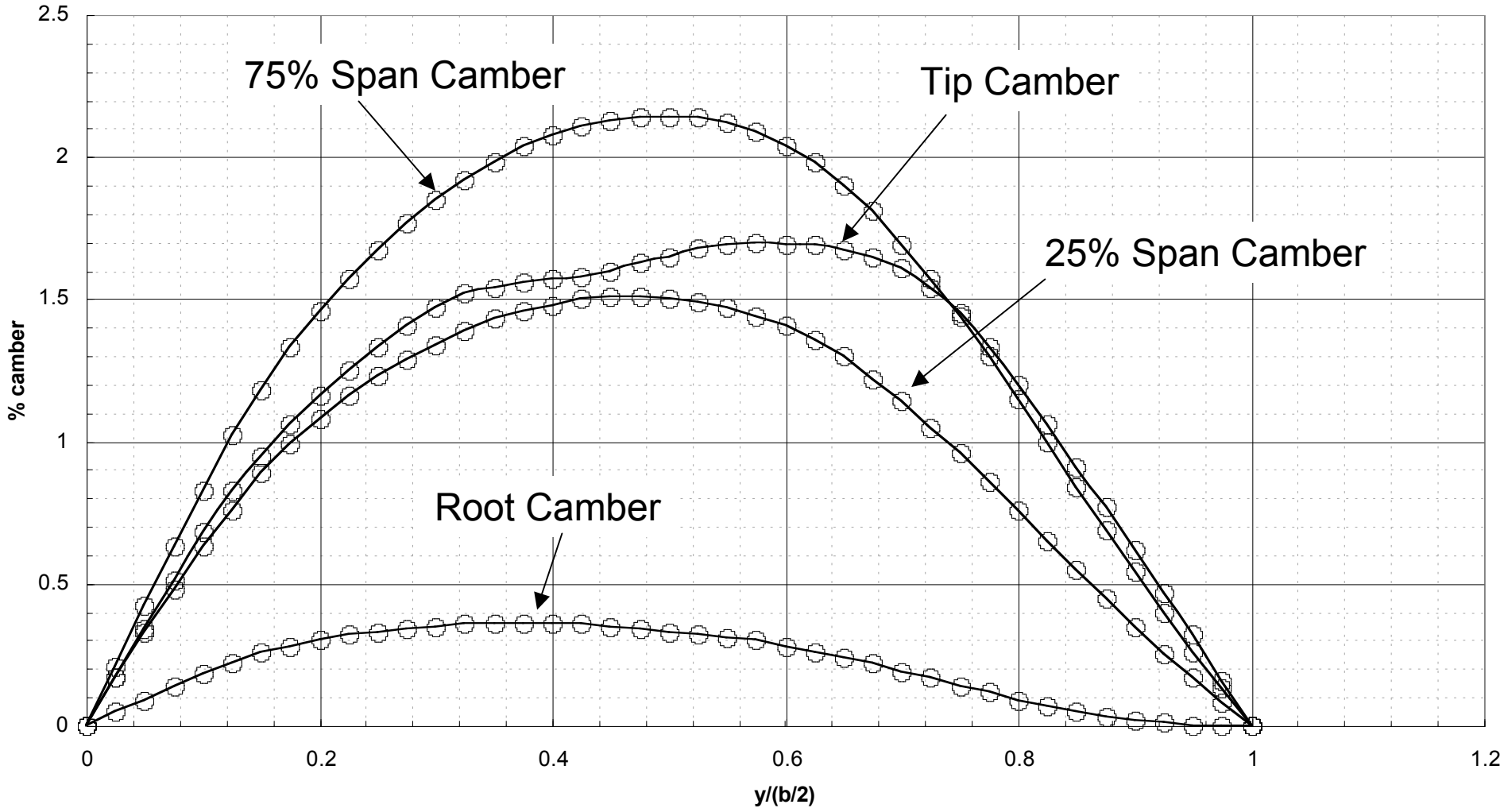
Twist Distribution for minimum drag at Cruise CL



Section CL Distribution



Root and Tip Mean Camber Lines

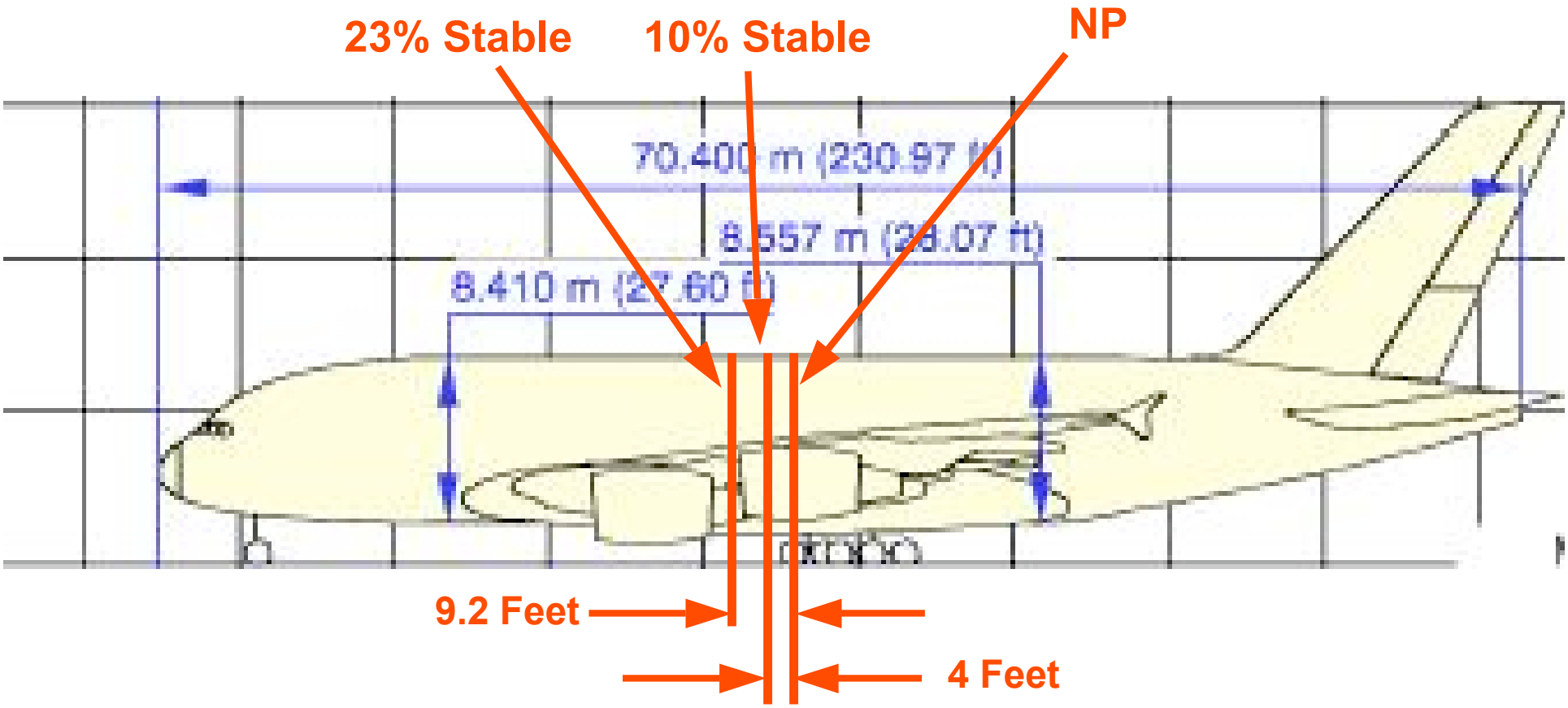


Stability

‘e’

NP – Neutral Point, aft CG limit for stability
(107.6 ft aft of LE of Fuselage)
10% Stable – CG 10% forward of NP
(fraction of mean chord)
23% Stable – LAMDES Minimum Drag Solution

‘e’ = 0.74



References

- Airbus Website
- UIUC Airfoil Database
- AIAA-2003-2886
  Commercial Aircraft
  - Software:
    - » LAMDES
    - » FRICTION
    - » VLMpc