Gate Box Requirement

A380  AR = 7.53
A340  AR = 9.21

\[
\frac{L}{D}_{\text{max}} \quad \text{is proportional to} \quad \sqrt{AR}
\]

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
alpha = 2 deg  M = .85

\( \chi/C \)

- 40% t/c  Cl = .963
- 70% t/c  Cl = .881
- 100% t/c  Cl = .685
alpha = 2 deg  M = 0.85

- 40% t/c
- 70% t/c
- 100% t/c
Surface Area Calculation

• Diagram used to model the A380 in CAD

• Complex bodies of revolution modeled in Inventor

• Planar bodies measured in AutoCAD
Calculation using FRICTION

Input Parameters

<table>
<thead>
<tr>
<th>Component</th>
<th>Wetted Area (ft²)</th>
<th>Reference Length (ft)</th>
<th>Thickness/Chord</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fuselage</td>
<td>16050</td>
<td>231</td>
<td>0.119</td>
</tr>
<tr>
<td>Wing</td>
<td>18284</td>
<td>41.5</td>
<td>0.082</td>
</tr>
<tr>
<td>Horizontal Tail</td>
<td>4211</td>
<td>22.3</td>
<td>0.081</td>
</tr>
<tr>
<td>Vertical Tail</td>
<td>2990</td>
<td>29.1</td>
<td>0.093</td>
</tr>
<tr>
<td>Engines (4)</td>
<td>3138 (each)</td>
<td>21.6</td>
<td>0.630</td>
</tr>
</tbody>
</table>

Reference Area: 9380 ft²
All surfaces assumed 100% turbulent flow
Wing, Horizontal, Vertical Tail - Modeled as Planer Surfaces
Fuselage, Engines – Modeled as Bodies of Revolution
Calculation using FRICTION

$C_{D_0}$

Results

<table>
<thead>
<tr>
<th>Altitude</th>
<th>Mach Number</th>
<th>$CD_0$</th>
<th>$CD_{min}$</th>
<th>$CL$</th>
<th>L/D Max</th>
</tr>
</thead>
<tbody>
<tr>
<td>5000</td>
<td>0.10</td>
<td>0.0212</td>
<td>0.0424</td>
<td>0.689</td>
<td>16.26</td>
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<tr>
<td>10000</td>
<td>0.50</td>
<td>0.0167</td>
<td>0.0334</td>
<td>0.611</td>
<td>18.32</td>
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<tr>
<td>20000</td>
<td>0.75</td>
<td>0.0161</td>
<td>0.0322</td>
<td>0.600</td>
<td>18.68</td>
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<tr>
<td>25000</td>
<td>0.80</td>
<td>0.0162</td>
<td>0.0324</td>
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<td>18.60</td>
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<tr>
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<td>0.85</td>
<td>0.0163</td>
<td>0.0326</td>
<td>0.605</td>
<td>18.51</td>
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<td>30000</td>
<td>0.89</td>
<td>0.0162</td>
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<td>18.65</td>
</tr>
</tbody>
</table>

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

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

Used to find
Minimum Drag CG Location
Minimum $C_L$
Twist Distribution
Section $C_l$ distribution
Root and Tip Camber
‘e’

Input
Same planform as FRICTION analysis
Mach = 0.85 (Cruise)
$C_D0 = 0.0163$
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

![Diagram showing twist distribution for minimum drag at cruise CL, with annotations for 'Tail' and 'Main Wing'.]
Section CL Distribution

![Graph showing Section CL Distribution with curves for Main Wing and Tail.](image-url)
Root and Tip Mean Camber Lines

75% Span Camber

25% Span Camber

Root Camber

Tip Camber

% camber

y/(b/2)
Stability

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

9.2 Feet
4 Feet
References

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