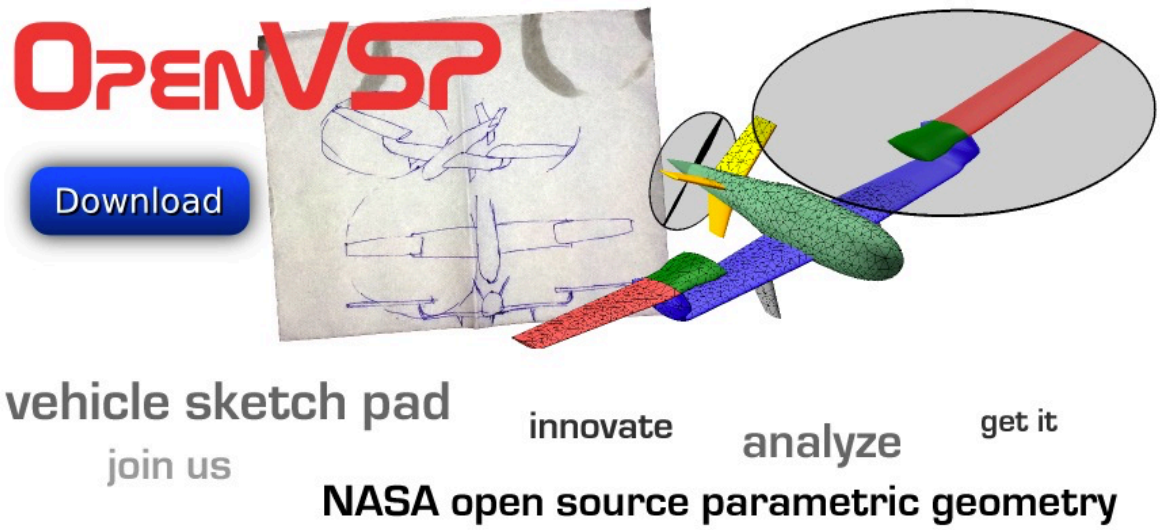
1



http://www.openvsp.org ☚ **Download executables**

http://www.openvsp.org/wiki ☚ **Online documentation**

https://groups.google.com/forum/#!forum/openvsp ☚ **Where to look when problems occur**

See also: Ex-youtube video files on Moodle in OpenVSP folder

2

OpenVSP – Vehicle Sketch Pad

A simple parametric modelling tool developed for aircraft-type geometries.

Warning: Open VSP is an open-source tool that is not completely reliable, or fully documented.

What it CAN do

1. Generate rendered/wireframe images suitable for conceptual design layouts
2. Export Step (.stp), IGES (.igs) and DXF files for use with other CAD packages
3. Simple/linear aerodynamic analysis using vortex lattice method - sufficient for aerodynamic load estimates, neutral point and trim calculations, induced drag estimation. Can estimate wetted areas.
4. Generate triangulated surface meshes for suitable for import into CFD packages
5. Centre of Gravity estimation by various means
6. Some limited internal layout
7. Handle simplified propellor/rotor aerodynamics using actuator disk model

What it CAN'T do – things better done using CAD/FEA/CFD packages

1. Generate dimensioned 3-view drawings
2. Show control surfaces, windows, doors, etc. (*Can* be done but other tools m
3. Handle detailed internal layout (same comment)
4. Perform structural layout or analysis
5. Estimate maximum C_L or handle high-lift systems, transonic – i.e. generally, nonlinear – aerodynamics

With new developments, more of these things are handled by OpenVSP...

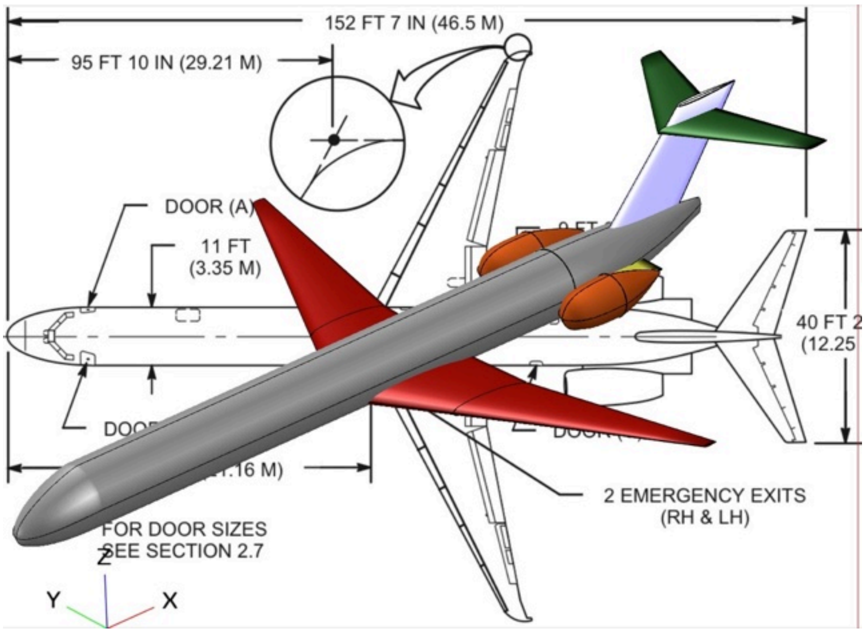
Workflow: OpenVSP → Concept Report and aero→ Solidworks (e.g.) → Final Report 3-view

3

Example: MD90 approximation

Example starting from existing 3-view of an MD90 (as in video tutorial 5).

For initial/conceptual design only a limited amount of detail is required – about the level one would include in a first-pass wind tunnel or CFD model. BUT – what is included should be accurate!



Analysis for the wing geometry for the model as built (not exact) gives:

S_{ref} = 124.0
c_{ref} = 4.71 (MAC, **external calculation***)
b_{ref} = 33.0.

These data are supplied as the first three lines of the XX_DegenGeom.vspaero file.

I also supplied α = AoA = 2°, also set the wing at an incidence angle of 1.5°, and used a NASA SC(2)-0714 wing airfoil.

The wing apex was at X = 0.558.

* OpenVSP does not (yet) calculate MAC. Use hand calculation, Mason's WingPlanAnal or other code.

4

VSPaero

1. *vspaero* is a vortex-lattice method (and, more recently, panel method) solver that will compute linear aerodynamics. Vortex lattice method (VLM) is the recommended choice.
2. It can find many useful things such as
   a. $dC_L/d\alpha$ for finite wings or whole vehicle
   b. Induced drag estimation
   c. Span loading (distribution of lift force along span)
   d. Stability derivatives and from this, location of aircraft Neutral Point and CG
   e. Trimmed C_L for a set CG location and rigging angles (find alpha that sets C_{my} = 0).

Online reference for *vspaero*

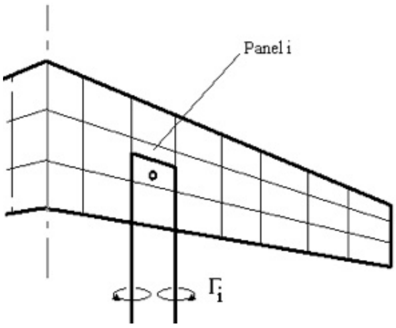
http://www.openvsp.org/wiki/doku.php?id=vspaerotutorial

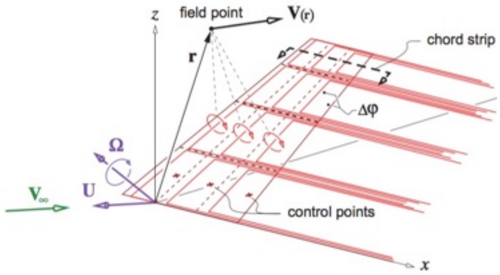
References for vortex lattice method (all available in library)

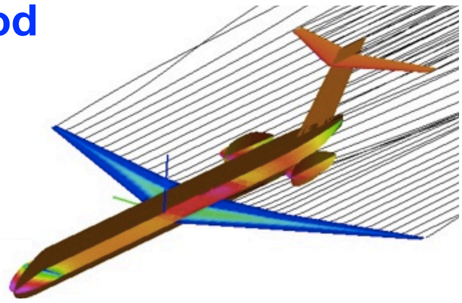
1. Sforza, "Commercial Airplane Design Principles" Butterworth Heinemann 2015, App C
2. Kuethe and Chow, "Foundations of Aerodynamics" 5e, Wiley, 1998
3. Drela, "Flight Vehicle Aerodynamics", MIT Press 2014, Chapters 2 and 6
4. Katz and Plotkin, "Low Speed Aerodynamics", 2e, Cambridge University Press 2001
5. Anderson, "Fundamentals of Aerodynamics" McGraw-Hill 2010, Chapter 5

5

The Vortex Lattice Method





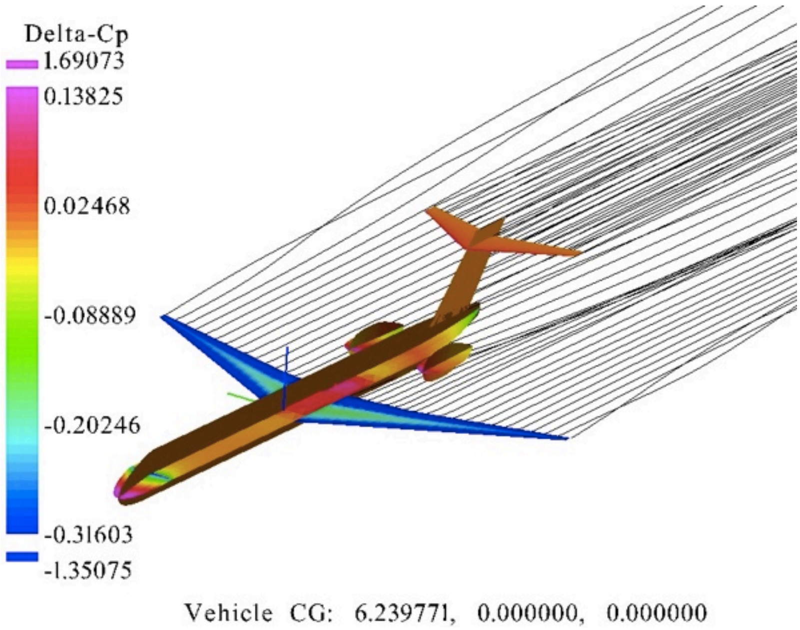


1. The vortex lattice method solves a discrete version of the linear potential flow equations. Unlike the lifting-line method, it can deal with lifting surfaces of arbitrary geometries. It assumes attached flow.
2. Lifting surfaces are idealised the same way as for thin airfoil theory: flow tangency (i.e. no-through-flow) is assumed with mid-surface slopes of each surface.
3. Each lifting surface is tessellated into a number of panels, and each of these is assigned a ‘horseshoe’ vortex Γ_i, typically bound at the 1/4 chord point of each panel. The strength of each such vortex is uniform along its length.
4. Every such panel is also assigned flow-tangency control point, typically at the 3/4 chord point of each panel (marked **o** in the above diagram).
5. There are then the same number of unknowns (vortex strengths) as equations (given by flow tangency at each of the control points) and a matrix problem can be set up and solved for the vortex strengths.
6. The positions of the trailing arms of each horseshoe vortex must be known to obtain a solution – typically these are (initially at least) assumed to be aligned with the direction of the free-stream velocity vector. For an iterative solution the trailing arm positions can be re-estimated after each iteration.
7. From the vortex strengths (and their positions), one can calculate overall lift forces, moments, and inviscid lift-induced drag. Note that we still need to independently calculate viscous (BL) drag.

6

VSPaero

Before getting started with more detailed analyses with the command-line tools, it is best to check the geometry is OK by making a solution using the *vspaero* GUI in *vsp*. (You may have to make a “degenerate geometry” before running the solver.) Check the computed ΔC_p distribution and vortex wakes look reasonable. If the solver takes a few hundred iterations to converge on each iteration, rather than *O*(20), there is likely something wrong.



Pressure coefficient contours and vortex wake arrangement for the MD90 model shown earlier.

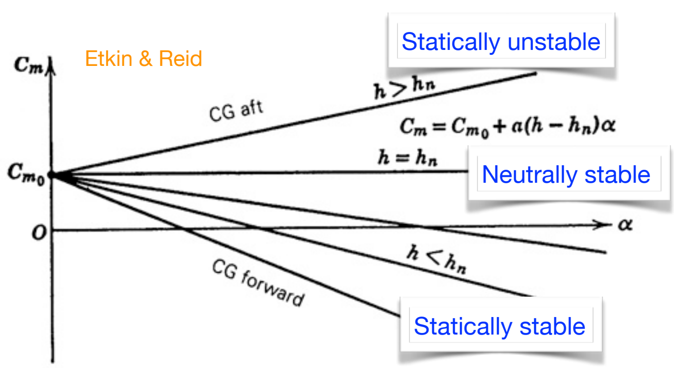
In order to get a reliable computation it proved necessary to delete from the model some stub pylons connecting engine pods to fuselage. Be prepared to experiment.

Note the “degenerate geometry” used in the analysis: round features such as fuselage and pods are represented by flat/cruciform approximations. This is good enough for preliminary aerodynamic estimates.

OpenVSP example: finding Neutral Point; setting CG location and wing and tail incidence/rigging angles

Computing the aircraft's Neutral Point and desired CG

By running *vspaero* from the command line with the "-stab" option (see online tutorial), we can get information from which we can find the aircraft's Neutral Point, and from this find the CG location that provides a given static margin of longitudinal stability.



Here h is the CG location given as a ratio of the wing's aerodynamic mean chord, MAC. Each line represents a different CG location.

The location of the Neutral Point is h_n and this gives a line with zero slope.

One output from the stability analysis is $dC_{m,y}/d\alpha$, i.e. the slope of these lines.

If we run the stability analysis with two different CG locations (as supplied by the .vspaero file), find the two different slopes $dC_{m,y}/d\alpha$, we can then use linear interpolation to find the CG location that gives zero slope (this *is* the Neutral Point, NP).

The value $h - h_n$ is the static margin for the aircraft. Typically/initially this is a (chosen) value of order 0.1. Hence to obtain a desired static margin we place the CG $(h_n - h) \times$ MAC forward of the neutral point. This is usually the rearmost permitted CG location.

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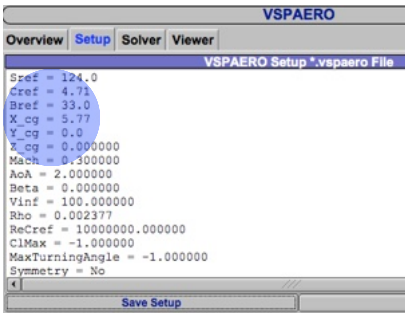
Setting MD90 Neutral Point and desired CG



We already had MAC c_{ref} = 4.71 (for which an external calculation based on wing geometry was required: see e.g. http://www.dept.aoe.vt.edu/~mason/Mason_f/MRsoft.html - WingPlanAnal).

The relevant data can be found in XX_DegenGeom.stab after running *vspaero* from the command line with the -stab option.

For AoA = 2.0 and X_cg = 0.0, find that C_L = 0.7662390 and $dC_{my}/d\alpha$ = –7.1730537/rad.

For AoA = 2.0 and X_cg = 5.0, find that C_L = 0.7662390 and $dC_{my}/d\alpha$ = –1.4252034/rad.

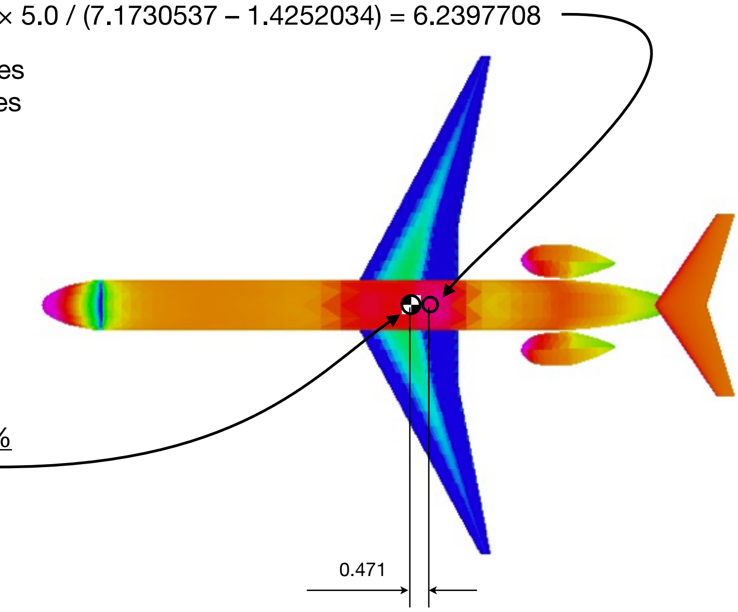
Negative values indicate positive longitudinal stability.

Hence X_cg(NP) = 5.0 + 1.4252034 × 5.0 / (7.1730537 – 1.4252034) = 6.2397708

(A check with X_cg = 6.2397708 gives $dC_{my}/d\alpha$ = 0, as required. This makes h_n = 6.2397708/4.71 = 1.3247921.)

(Note: in recent versions of VSPaero, stability calculations can be run directly from the GUI. The neutral point - NP - is reported right at the send of the resulting .stab file.)

<u>So to achieve a static margin of 10% we would place the CG at X = 6.2397708 – 4.71 × 0.1 = 5.77.</u>

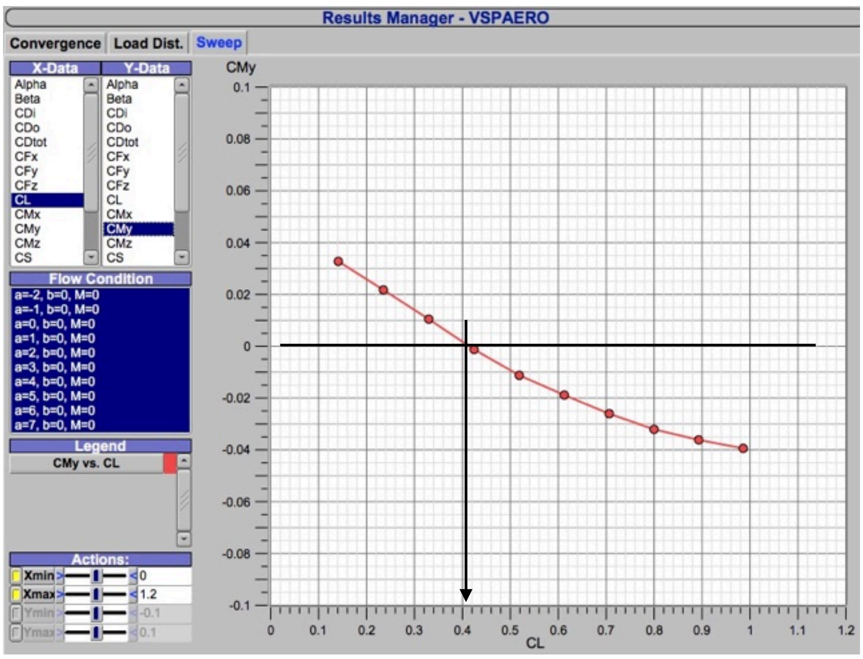


10

Finding/setting rigging angles for trimmed C_L

With the CG in its required location and with the washout distribution set, the wing and tail rigging angles need to be set to bring the aircraft into trim (C_{my} = 0) for the design (generally, cruise) C_L. At the cruise C_L, the fuselage and other non-lifting components should align with the free-stream (i.e. the wing and tail rigging should bring the aircraft to the cruise C_L with C_{my} = 0, all at AoA = 0). This can be approximately done by trial and error (changing wing and tail rigging angles) and using a vspaero Sweep plot of C_{my} vs C_L.

In this example, we see that the trimmed $C_L \approx 0.41$. This is achieved at $\alpha \approx 0.9°$.

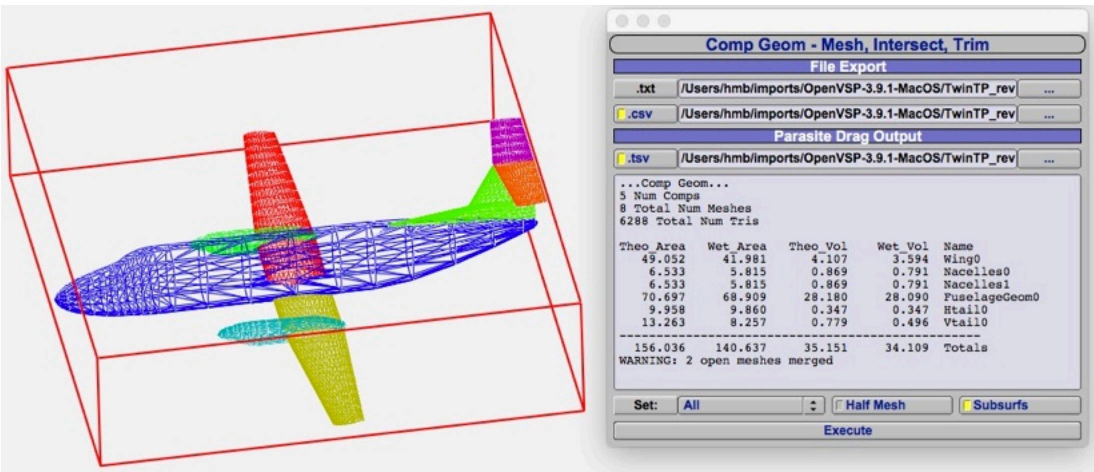


OpenVSP example: *vspaero*-aided drag polar estimation

12

Twin Turboprop: 2-term drag polar with aid of *vspaero*

Construct a 2-term drag polar model for the twin turboprop already developed in class, using skin friction estimates from VSPaero for non-lifting surfaces and tails, experimental data for wing airfoil profile drag, and VSPaero's vortex lattice solver for inviscid lift-dependent drag.



Use the Comp Geom tool to get wetted areas of the major components.

Under the Wing Planform tab of the Geometry Browser, find key reference data. (But note: the chord here is NOT the MAC. That needs to be separately computed, e.g. using Mason's WingPlanAnal code. For the current geometry, MAC = 1.490 m.)

Twin Turboprop: 2-term drag polar with aid of *vspaero*

Run vspaero. Outputs relevant to drag estimation appear in the XXmodelXX_DegenGeom.history file.

Non-lifting viscous drag (taken to include tail surface drag)

Though the skin friction contributions to $C_{D,0}$ are not verified correct, we'll adopt them here to illustrate the methodology.

```
Skin Friction Drag Break Out:


Surface                                     CDo

Wing,0                                     0.00475
Wing,1                                     0.00475
Htail,0                                    0.00104
Htail,1                                    0.00104
Vtail,0                                    0.00196
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
Nacelles                                   0.00035
FuselageGeom                               0.00345
FuselageGeom                               0.00345
FuselageGeom                               0.00345
FuselageGeom                               0.00345
```

These skin friction estimates are independent of angle of attack (and C_L).

We'll use vspaero's skin friction drag estimate for everything but the wing's contribution (for which we'll adopt airfoil test data for profile drag). The total contribution is $C_{D,0,nl}$ = 2 x 0.00104 + 0.00196 + 8 x 0.00035 + 4 x 0.00345 = 0.02064.

Twin Turboprop: 2-term drag polar with aid of *vspaero*

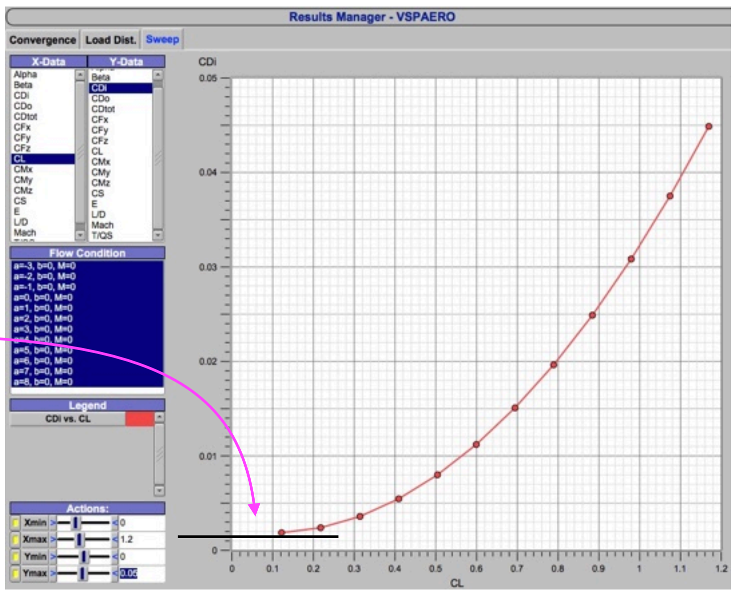
Run vspaero. Outputs relevant to drag estimation appear in the XXmodelXX_DegenGeom.history file.

Inviscid induced drag

Run vspaero over a range of angles of attack that generate moderate (and positive) C_L values. Note the C_{Di} values in the .history file. Best to run in Batch mode so that outputs for all angles of attack are catenated to the .history file. The values can be plotted in the Sweep window.

Note that C_{Di} does not asymptote to zero at C_L = 0.

That's because at overall C_L = 0,
(a) owing to the wing's washout distribution various parts of it lift either up, or down, generating induced drag, and
(b) the horizontal tail has negative lift, hence generates its own induced/ trim drag.



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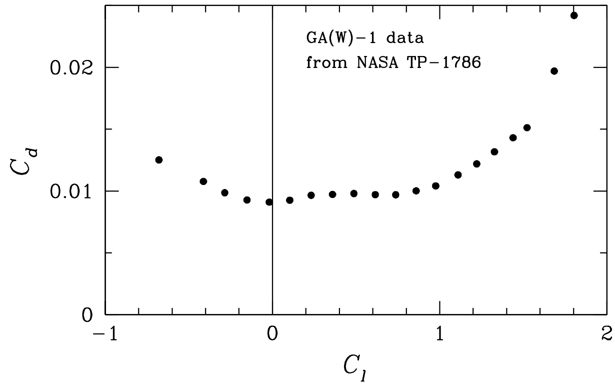
Twin Turboprop: 2-term drag polar with aid of *vspaero*

Airfoil profile drag

This can be either generated using a 2D viscous–inviscid interaction code (e.g. XFOIL), or by adopting wind tunnel test data, if any. I've used the NASA LS GA(W)-1 airfoil series for the wing, with test data available from the NASA Technical Reports Server: NASA-TM-78709.



These are the data for the 17% thick section used at the wing root.

(The Reynolds number should be checked...)

We'll assume the polar is representative of the whole wing, even though the wing transitions to a 13% thick section at the tips.

```
...Comp Geom...
5 Num Comps
8 Total Num Meshes
6288 Total Num Tris

Theo_Area   Wet_Area   Theo_Vol    Wet_Vol  Name
  49.052     41.981      4.107      3.594  Wing0
   6.533      5.815      0.869      0.791  Nacelles0
   6.533      5.815      0.869      0.791  Nacelles1
  70.697     68.909     28.180     28.090  FuselageGeom0
   9.958      9.860      0.347      0.347  Htail0
  13.263      8.257      0.779      0.496  Vtail0
------------------------------------------------------
 156.036    140.637     35.151     34.109  Totals
WARNING: 2 open meshes merged
```

We will only use these data for the exposed wetted area of the wing. Comparing this to its total wetted area, we see that we have to factor the profile drag contribution by approximately

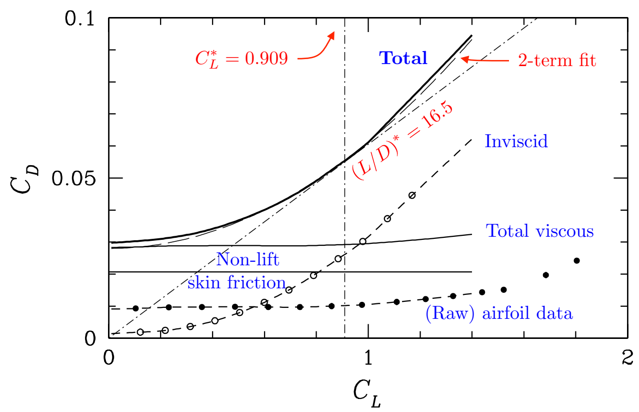
41.981/49.052 = 0.8445.

16

Twin Turboprop: 2-term drag polar with aid of *vspaero*

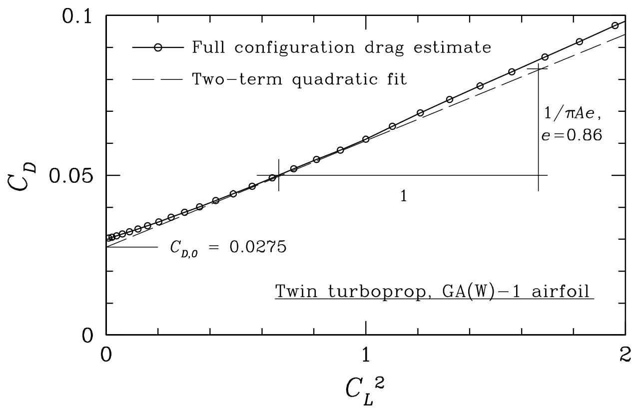
Sum up the contributions

$C_D = C_{D0,nl} + 0.8445 \times C_d(C_L) + C_{Di}(C_L)$



Plot C_D vs C_L^2,

fit a 2-term quadratic over likely range of C_L.



Finally, our refined estimates:
$C_{D,0} = 0.0275$
$e = 0.86$
giving $(C_L/C_D)_{max} = 16.5$

cf. the values initially adopted from texts:
$C_{D,0} = 0.02$
$e = 0.80$
and $(C_L/C_D)_{max} = 18.6$

OpenVSP example: Setting wing twist distribution.

Setting wing washout distribution

For any wing planform, two big aerodynamic issues are

1. Minimising induced drag, especially near cruise C_L – this is an aerodynamic efficiency issue.
2. Avoiding initiation of stall near the wing tips (a.k.a. tip stall) at high C_L – this is a handling issue.

For unswept wings, we know an elliptical chord distribution achieves the first of these things at all C_L, without wing twist. And in the absence of *Re* effects, no location along the span is worst for initiation of stall in that case. However, most production wings are made using trapezoidal sections. In this case we can twist the wing to achieve either of the goal above. With some care (and perhaps varying airfoil sections along the span, as well) we can come close on both goals.

We need to make the distinction between the spanwise lift (or loading) distribution $\Gamma(y) = \rho V c(y) C_l(y)$ and the spanwise coefficient of lift distribution $C_l(y)$. In considering induced drag we are concerned with $c(y)C_l(y)$; in considering tip stall we are concerned with just $C_l(y)$.

It is useful to have the idea of the ***basic*** and ***additional*** lift distributions. The *basic* distribution is for $C_L = 0$ with the twisted wing, while the *additional* lift is what one would get at different angles of attack with an untwisted wing. At small angles the superposition is linear.

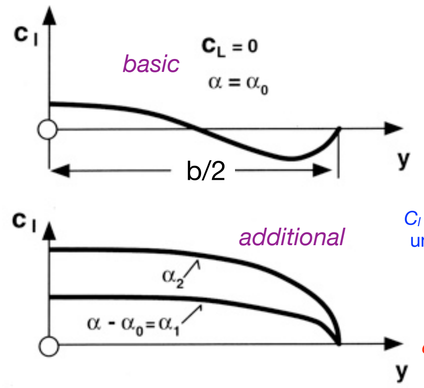


Fig. 55: Lift distribution on twisted wing.
- basic lift distribution for $\alpha = \alpha_0$
- additional lift due to angle of attack $\alpha - \alpha_0$

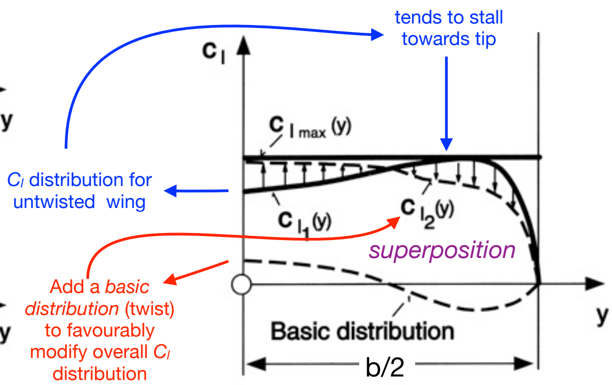
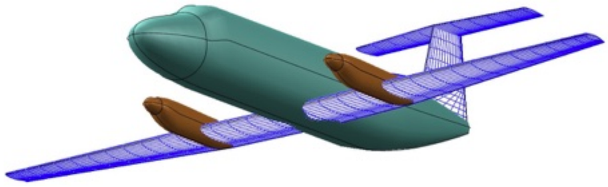


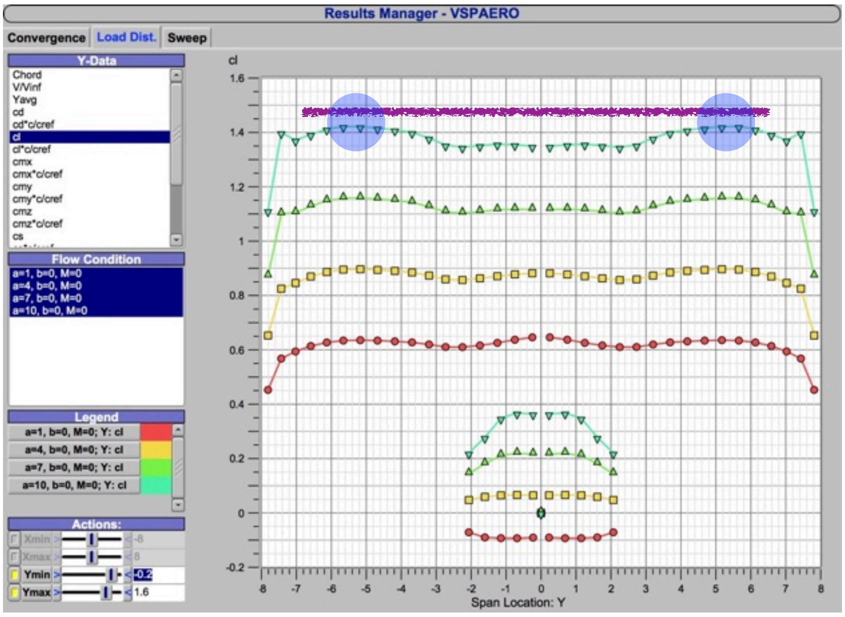
Fig. 56: Improved stall characteristics through superposition of an appropriate basic lift distribution.

Thomas - Fundamentals of Sailplane Design

Setting wing washout distribution



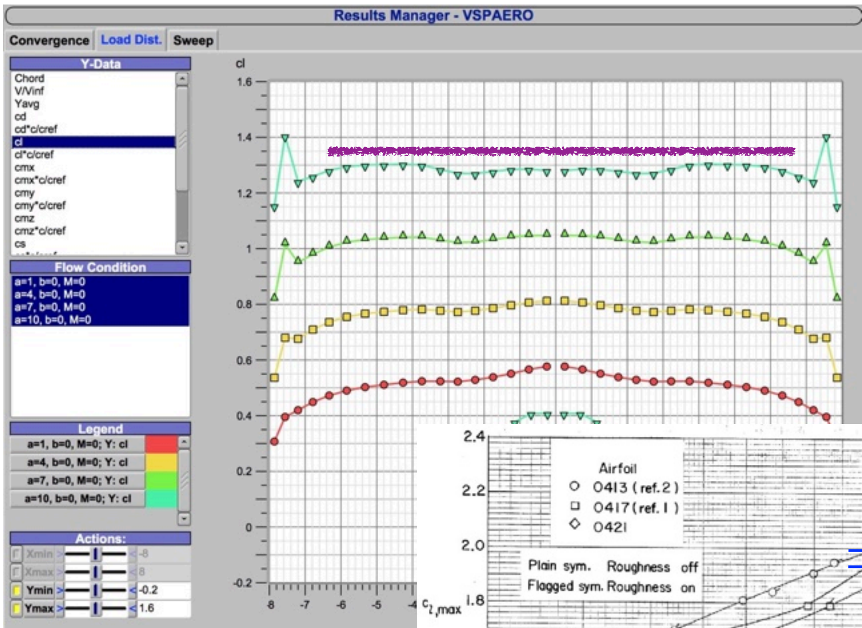
Shown below are the C_l distributions for a Twin Turboprop with GA(W)-1 airfoils and no wing twist. At high angles of attack (high overall values of C_L) we can see that stall is liable to initiate on the outboard wing panels.

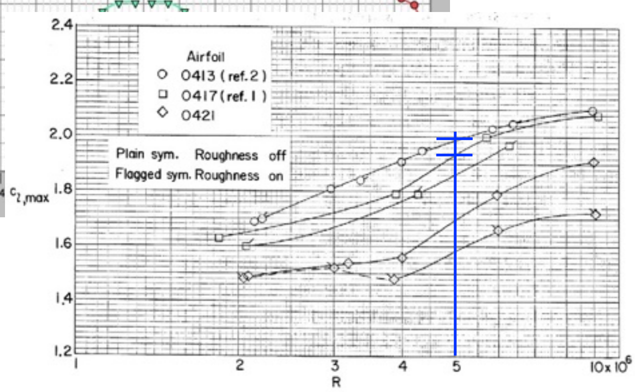


So next we add some washout, progressing to 2.5° at the wing tips, to change the basic C_l distribution.

Setting wing washout distribution

Adding washout at the wing tips can significantly improve the C_l distribution (but not necessarily the lift distribution, which is proportional to $c*C_l$). The differences in maximum C_l are now minor enough that we could likely ensure the tips do not stall first by changing the airfoil section towards the tips (to an airfoil that has a higher geometric stall angle of attack). NOTE: it is generally a poor idea to overdo washout. The example below is for a tip washout of 2.0°.





Approach speed Reynolds number of order 5x10⁶ in wing tip region. According to wing tunnel data (right), $C_{l,max}$ for 13% thick tip airfoil is greater than 17% thick root airfoil by approx. ratio 2/1.95 which is just enough to make stall initiate at wing root (depending on eventual stall AoA). Marginal but probably OK.

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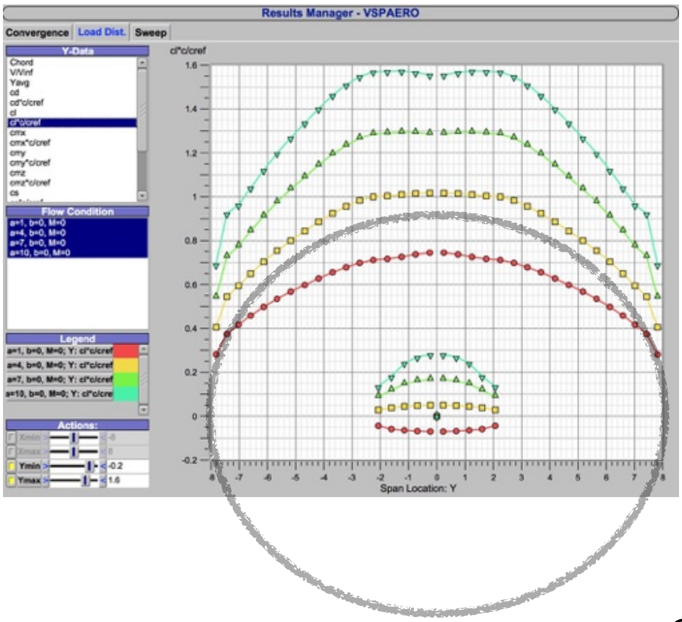
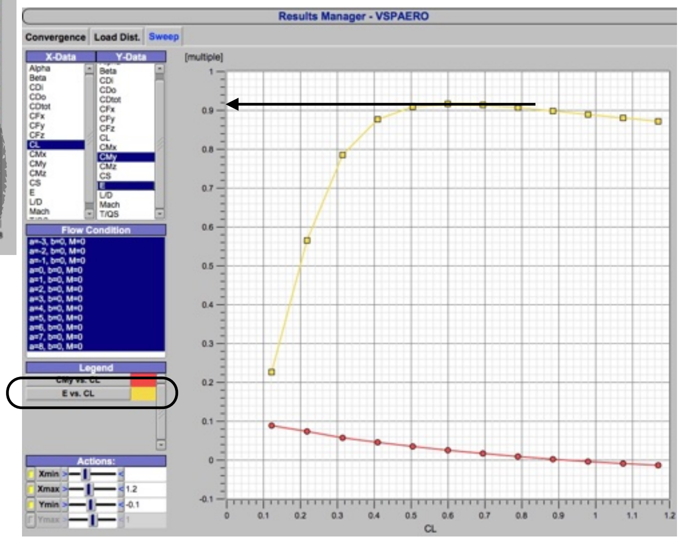
Check inviscid aircraft efficiency *e*

If we examine the wing's lift load distribution $\Gamma(y)$, proportional to $c \times C_l$, we find it most closely approaches an elliptical shape (the theoretical optimum) at an aircraft angle of attack of around 3 degrees. This AoA would give an overall C_L of around 0.6, which is about where the inviscid aircraft efficiency peaks at a maximum of around 0.92.



Note that the loading distribution is affected by changes to washout distribution. **In fact, it is theoretically possible to exactly achieve an optimal/elliptical overall loading distribution (though only at one value of C_L) for any wing planform by changing the washout distribution (i.e. the basic lift distribution)**. However, that may conflict with what is required to avoid tip stall, which probably is a more important consideration. Ultimately the designer has to adjust planform, washout and airfoil choices to achieve the most acceptable overall solution.



The value of *e* reported by VSPaero is only based on inviscid induced drag. In reality, there will also be viscous lift-induced drag, making the true overall value of *e* somewhat lower.

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OpenVSP example: aero design for a swept wing

23

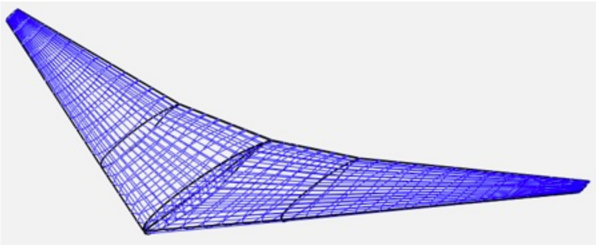
Wing analysis using *vspaero*

Preliminary analysis and optimization of wing performance can be carried out with *vspaero* – enough to get the geometry into the right ballpark. It is best to deal with the wing on its own initially - delete other parts.

The main issues to be dealt with are balancing low-speed (stall) performance with cruise performance (induced drag minimization). A first pass at these two things can be made with the linear aero tools that *vspaero* provides.

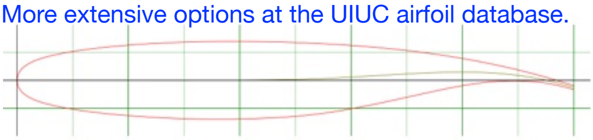
The tradeoffs are especially demanding for swept wings – and these cannot adequately be dealt with using Prandtl's lifting line theory. A numerical tool such as *vspaero* is the only real option.

We will take the wing planform geometry as given, choose an airfoil family, and vary the wing's thickness and twist distribution to balance stall and induced drag performance.



Choose an airfoil family. For swept jet transport aircraft, 'supercritical' sections are the norm. I used a NASA SC(2)-0714 which is 14% thick and has design (cruise) C_l = 0.7. (Neither the exact airfoil nor its thickness are critical at this stage, but a reasonably thick airfoil is to be preferred.)

I went to www.airfoiltools.com, put in 'supercritical' as a search term, downloaded a .dat file for the chosen airfoil. More extensive options at the UIUC airfoil database.



Load this airfoil into VSP under the Wing/Airfoil tab. Initially at least, all sections 0, 1, 2 (root through tip).

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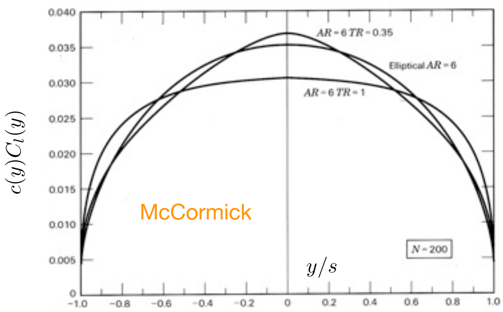
Spanwise lift distribution

The spanwise lift distribution $c(y) \times C_l(y)$ is proportional to the circulation distribution $\Gamma(y)$ and tells us how lift force per unit length is distributed along the span. Integral is prop. to the wing's total lift coefficient C_L.

We can also use $\Gamma(y)$ to compute the wing's shear force and bending moment diagram.

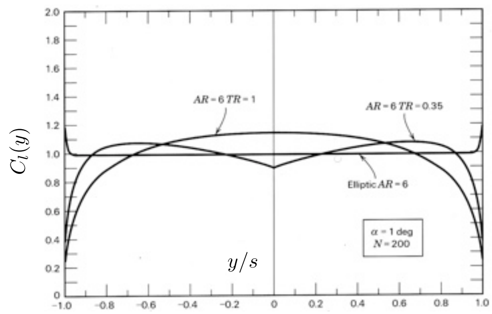
Recall that according to lifting line theory (developed for unswept wings) $\Gamma(y)$ should be as close as possible to an elliptical distribution in order to maximize span efficiency e (minimize induced drag).



$$2\Gamma(y)/V_\infty = c(y)\,C_l(y)$$

From $\Gamma(y)$ we can work back and estimate the local C_l value (and how close it is to $C_{l,max}$).

Stall is predicted to initiate at the spanwise location where C_l first reaches $C_{l,max}$.



Very often, twist (typically, 'washout': airfoil twisted nose-down) is introduced in order to influence $C_l(y)$. In turn this alters $\Gamma(y)$ and hence the wing's span efficiency.

Adding twist also makes the shape of $\Gamma(y)$ a function of overall C_L (or α). The zero-C_L $\Gamma(y)$ is called the 'basic' lift distribution, and the remainder from the total $\Gamma(y)$ at any particular C_L is the 'additional' distribution.

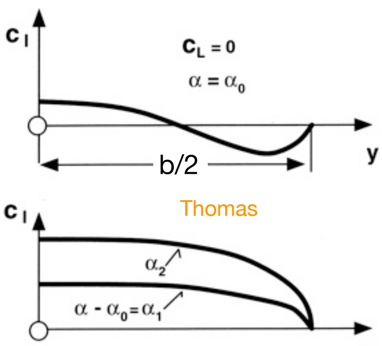


Fig. 55: Lift distribution on twisted wing.

- basic lift distribution for $\alpha = \alpha_0$
- additional lift due to angle of attack $\alpha - \alpha_0$

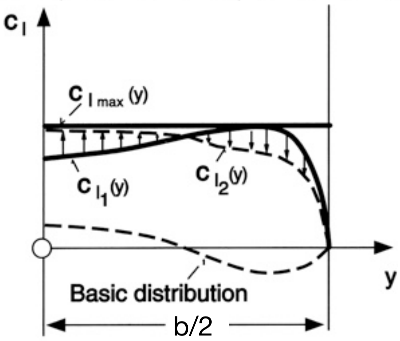


Fig. 56: Improved stall characteristics through superposition of an appropriate basic lift distribution.

Swept wing subsonic aerodynamics

The basic reason for choosing a swept wing is to delay the Mach number for onset of transonic drag rise.

However, introducing sweep has a number of undesirable aerodynamic (not to mention structural) side effects. Compared to an unswept wing of the same aspect ratio, sweep

1. Alters the spanwise circulation distribution to lower the span efficiency (increases induced drag);
2. Associated with this, relatively increases the loading on the wing tips compared to the root, promoting tip stall;
3. Decreases the lift curve slope of the wing.

To mitigate these effects, the designer chooses the least sweep that will acceptably delay transonic drag rise and alters geometry (chord, twist and airfoil distributions along the span) to achieve an acceptable compromise between the first two of these items.

Reminder: we are taking the planform (span, area, sweep and chord distribution) as fixed parameters. That leaves us with airfoil section (chiefly, its thickness, in order to change the local maximum C_l) and twist (a.k.a. washout) distributions along the span as design variables.

Problem: twisting the wing to avoid stall at the tips changes the spanwise lift distribution and may lower the span efficiency at some angles of attack. It may also make the tips produce negative lift at low C_L (i.e. high speed).

Problem: Making the airfoil thinner near the root (to decrease sectional C_{lmax} and hence promote root stalling first) ultimately lowers the wing's overall maximum C_L capability, reduces wing volume (fuel capacity) and strength (or for same strength, increases the weight). Conversely, increasing tip thickness, while reducing susceptibility to tip stall, lowers the drag-divergence Mach number.

The aerodynamic design engineer's task is to find an acceptable compromise between the conflicting requirements.

Swept wing subsonic aerodynamics

1

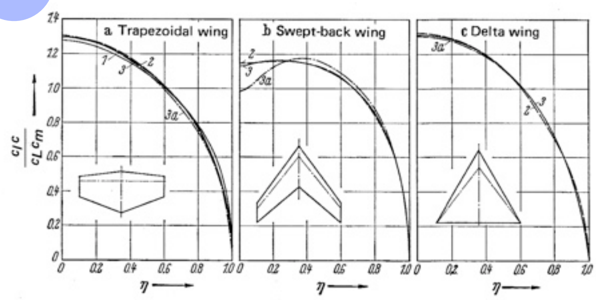


Figure 3-36 Lift distribution $c_l c/c_L c_m$ of three wings without twist of Table 3-5 and Fig. 3-33, $c'_{L\infty} = 2\pi$; $c_m = A/b$ = mean wing chord. Curve 1, simple lifting-line theory of Multhopp. Curve 2, extended lifting-line theory of Weissinger. Curve 3, lifting-surface theory of Truckenbrodt. Curve 3a, lifting-surface theory of Wagner (five-chord distributions).

2

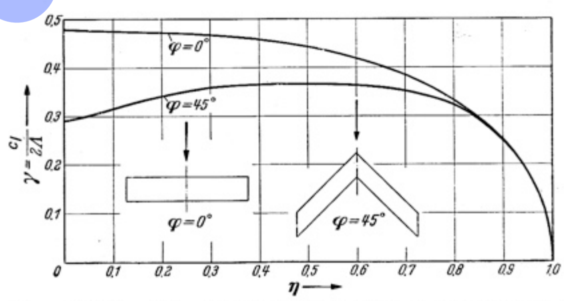


Figure 3-47 Circulation distribution and distribution of the local lift coefficients over the span for two wings of constant chord; aspect ratio $\Lambda = 5$, sweepback $\varphi = 0$ and $\varphi = 45°$; $c'_{L\infty} = 2\pi$; $\alpha = 1$; lifting-surface theory of Truckenbrodt [84].

Compared to an unswept wing of the same aspect ratio, sweep

1. Alters the spanwise circulation distribution to lower the span efficiency (increases induced drag);
2. Associated with this, relatively increases the loading on the wing tips compared to the root, promoting tip stall;
3. Decreases the lift curve slope of the wing.

3

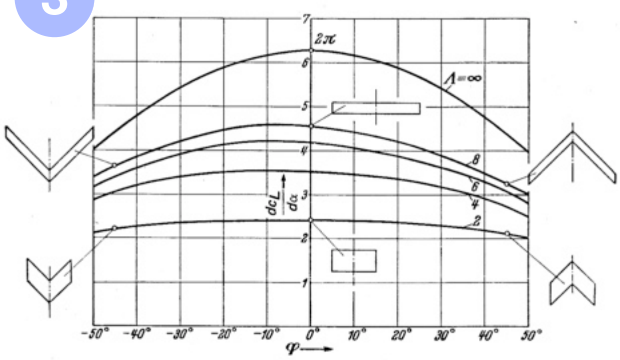


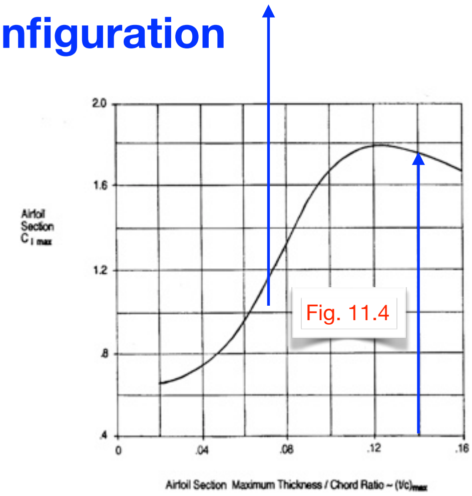
Figure 3-46 Lift slope of swept-back wings of constant chord vs. sweepback angle φ and aspect ratio Λ, from [103]; extended lifting-line theory. Curve for $\Lambda = \infty$: $\cos\varphi$ law from Eq. (3-123).

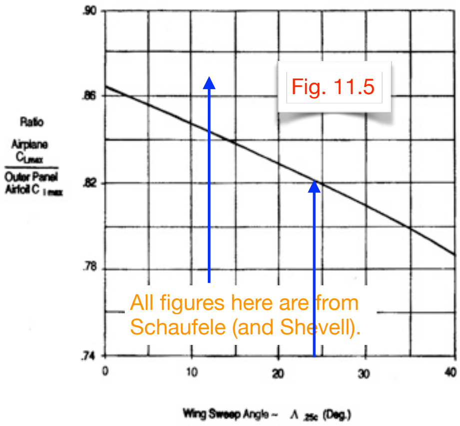
All figures here from Aerodynamics of the Airplane by Schlichting & Truckenbrodt.

Tip stall mitigation, clean configuration

Process (assuming sweep and basic airfoil have been chosen)

1. Estimate the tip panel airfoil $C_{l,max}$ based on the chosen basic airfoil and Fig. 11.4.
2. Estimate corresponding wing design overall $C_{L,max}$ based on Fig. 11.5 and basic wing sweep at $c/4$.
3. At an appropriate overall wing angle of attack required to achieve this value of $C_{L,max}$, choose a wing twist distribution *and* thickness distribution such that stall initiates well towards the wing root, as shown in Fig. 11.2.
4. The wing thickness distribution is not simple to change in VSPaero unless we have .dat files for a family of related airfoils*. For now we will ignore this. Reduce the wing angle of attack to place overall wing C_L at the design cruise value and compute span loading and span efficiency.
5. If span efficiency is too low, be prepared to iterate!





The final wing cruise angle of attack and twist distribution are inserted into the whole-aircraft .vsp3 file for computation of the neutral point and CG location.

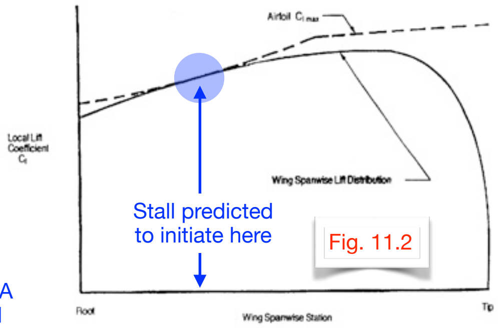


Fig.11-2 Span Loading/Airfoil $C_{l_{max}}$ Relationship for Proper Stall

* Airfoil .dat files for all the NASA SC(2) supercritical foils detailed in NASA TP-2969 can be found at http://m-selig.ae.illinois.edu/ads/coord_database.html#N

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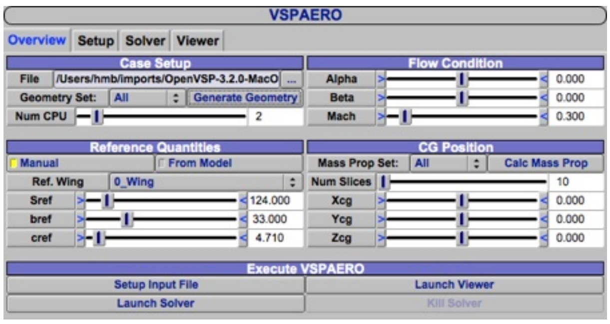
Using *vspaero* for wing performance analysis

Read

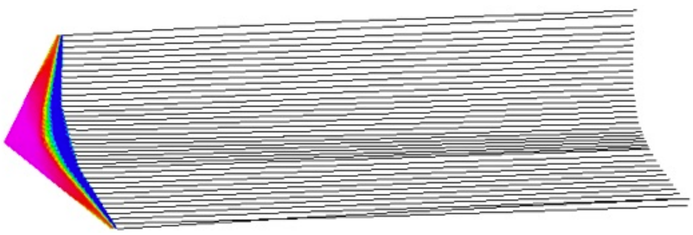
http://www.openvsp.org/wiki/doku.php?id=vspaerotutorial

We have already set the wing planform and (initial) airfoil choice, and the wing is (so far) untwisted.

The process for using *vspaero* to perform wing performance analysis using the GUI is



1. Create a 'degenerate geometry' using Analysis --> DegenGeom or VSPAERO --> GenerateGeometry to write out files XX_DegenGeom.csv, and .m;
2. Set Reference Quantities and Flow Condition under VSPAERO menu (note you should have the wing area *Sref*, span *bref* and MAC *cref* precomputed. Then 'Setup Input File' to write out XX_DegenGeom.vspaero;
3. Launch Solver. When it has terminated you may want to check the solution using LaunchViewer --> Aero --> Delta-Cp and Wakes;
4. Examine the XX_DegenGeom.history file to see Span Efficiency (*E*) and *CL, CDi* (you are advised not to trust the estimates of CDo and CDtot here). Examine/plot the XX_DegenGeom.lod file to see the distribution of c and C_l. Compute and plot the lift distribution ($c \times C_l$ vs y).



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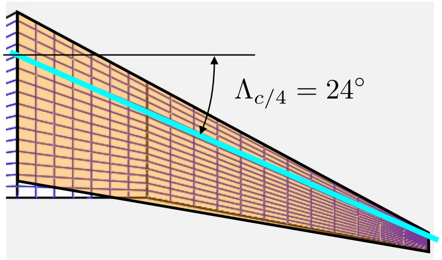
Using *vspaero* for wing performance analysis

Note the approximation by a simple trapezoid to estimate sweep angle.



Relationship between local circulation and the product of local chord and C_l:

$$\Gamma(y) = \tfrac{1}{2} V_\infty c(y) C_l(y)$$

Now, if we plot the ratio $\dfrac{c(y)C_l(y)}{c_g C_L}$

using the geometric mean chord $c_g = S/b$

we can always compare this to the normalised elliptical (optimal) distribution

$$\frac{2\Gamma_0}{V_\infty c_g C_L}\sqrt{1-\left(\frac{y}{b/2}\right)^2} = \frac{4}{\pi}\sqrt{1-\left(\frac{y}{b/2}\right)^2}$$

which gives span efficiency e = 1.

This comparison gives us some indication of efficient ways in which to vary the twist distribution.

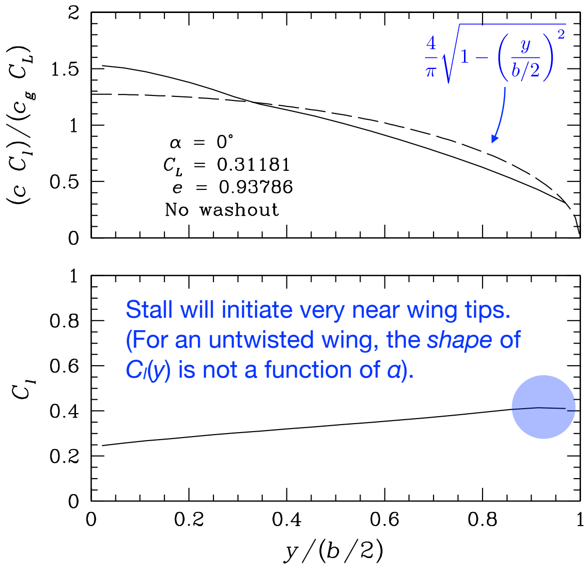
Using Schaufele's correlations:

Based on our assumed 14% tip airfoil thickness (probably too large), sectional maximum C_l = 1.75.

Based on c/4 sweep, (clean airplane $C_{L,\max}$)/(tip $C_{l,\max}$) = 0.82.
So the estimated clean $C_{L,\max}$ = 1.75 × 0.82 = 1.43.

Back to *vspaero*:

For orientation, below we show the loading distribution for the untwisted wing at zero AoA, α = 0°, calculated using *vspaero.*



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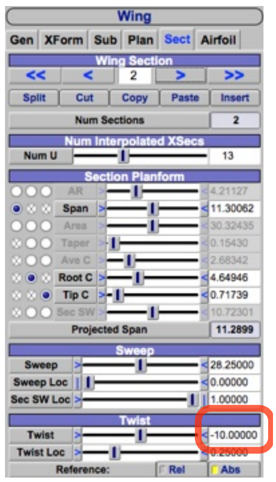
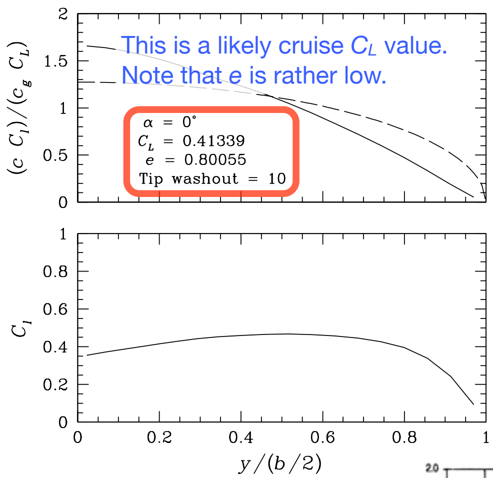
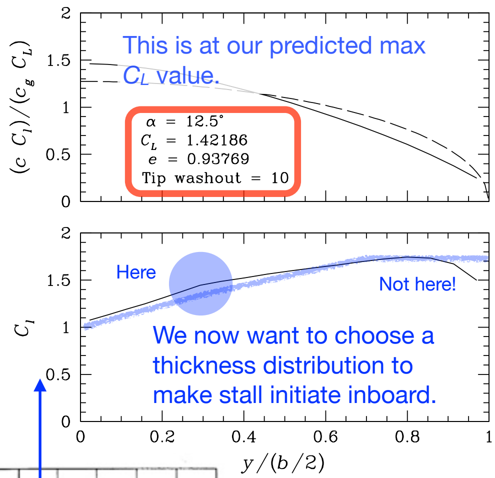
Using *vspaero* for wing performance



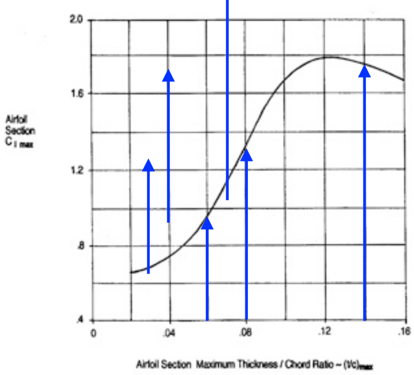
As a first pass, choose a linear tip washout of 10° and leave the centre panel untwisted.

After a few computations at different α, we find α = 12.5° gives C_L = 1.42.





Examination of Schaufele's correlation suggests that making the airfoil 8% thick at the taper break and 6% thick (or maybe a bit less) at the wing root could be acceptable.



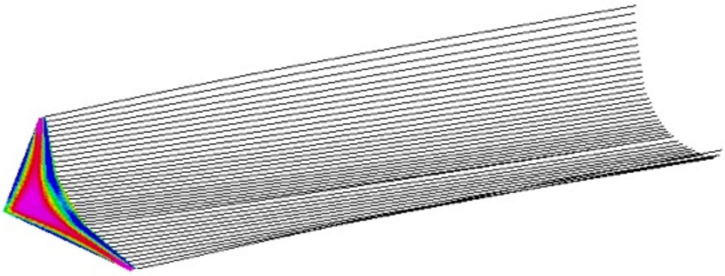
So now we can install those sections and re-run the analysis.

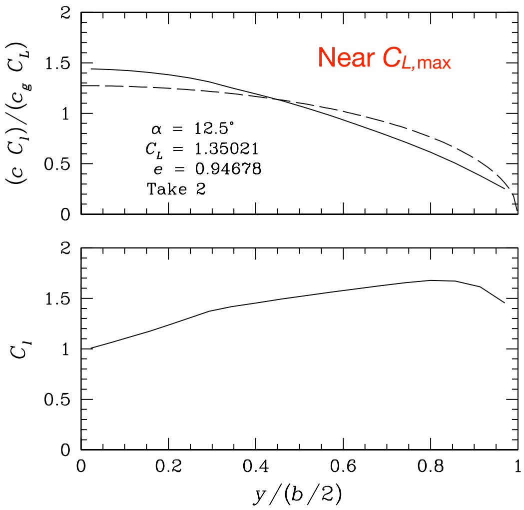
In fact in the NASA SC(2) family, we can only locate 4%, 6%, 10% and 14% thick foils: We will make do with 6%, 10%, 14% progression in our analysis and assume the airfoils can be designed.

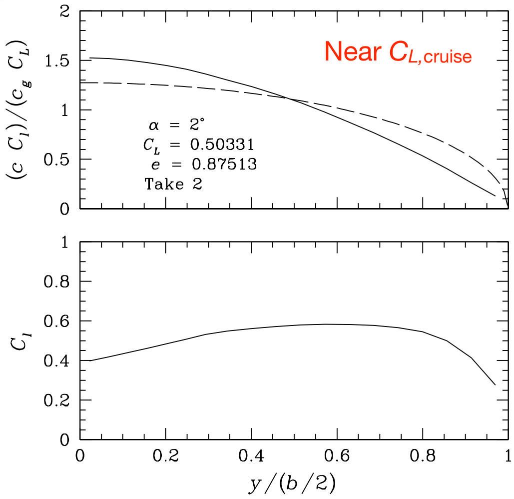
We also note a concern that our cruise span efficiency may be unacceptably low.

Using *vspaero* for wing performance analysis

Here are the outcomes of analysis at α = 12.5° (assumed $C_{L,\max}$) and α = 2° (assumed cruise C_L = 0.5) with the new airfoils, still with 10° washout at end of tip panels.







We note that the cruise value of span efficiency has now picked up to 87.5%, which is possibly acceptable. Next we move on to stability analysis and setting the CG position for a required static stability margin.