Detailed Considerations in the Stressing of an Aircraft Fuselage
Background

• Over 25 years experience designing and analysing high performance structures
• Stress Analysis in Advance Composites
• Certification Validation

• Design and Manufacture the first, wholly in the UK, brand new certified aircraft for over two decades.
• Swift-VLA is a fully composite, aerobatic two seated monoplane
OVERVIEW

1. Challenges of Composite Fuselage Design
2. NCAMP Design Allowables
3. Preliminary Design Method
4. Producing GFEM to confirm initial design
5. Determination of Critical Load cases
6. Applying Critical Balanced Load cases to GFEM
7. GFEM results and weight saving
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Why composites?

- Strength/weight ratio
- Formability
- Fatigue resistance
What are the challenges?

• Orthotropic nature of composite materials

• Generation of Design allowables for advanced composites still proves to be costly and time consuming.

• Extensive coupon test plans required to satisfy regulations.

• Manufacturers are unwilling to share design allowables generated.

• Leads to consecutive test programs and design allowables being generated by multiple aircraft manufacturers at great cost to each.
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Solution

- In an attempt to eliminate this waste within industry, the National Centre for Advanced Materials Performance (NCAMP) was created in the USA.

- Aim to populate a shared database of fully qualified material systems that will be freely available to the public.

- Allow manufacturers to gain certification by using a shorter equivalency test program, instead of having to conduct a full qualification program.

http://www.niar.wichita.edu/coe/ncamp.asp
EASA and FAA acceptance

Both the Federal Aviation Administration (FAA) the European Aviation Safety Agency (EASA) accept composite specification and design values developed using the NCAMP process.

• Acceptance detailed in –

  EASA Certification Memorandum CM-S-004
  FAA Policy Memorandum AIR100-2010-120-003
Current NCAMP qualified Material Systems

• Hexcel 8552
• Newport NCT4708
• Cytec MTM45-1
• Cytec 5320-1
• Park Aircraft E752 (coming soon)
• Tencate TC250 (coming soon)

• Each system has a range of product forms available, including various unidirectional tapes, plain weave fabrics and satin weave fabrics using both carbon and glass fibres.
Material System Specifications

• Each qualified material system comes with an associated NCAMP
  • Material Specification
  • Process Specification
  • Statistical Analysis Report
In the Statistical Analysis Report NCAMP provides Design Allowables for both LAMINA and various LAMINATES for multiple environmental conditions, ranging from cold dry (CTD) to hot wet (ETW2).

- Laminate layups tested

<table>
<thead>
<tr>
<th>Layup Ply Orientation (%0, %±45, % 90 degrees plies)</th>
<th>Layup Description</th>
<th>Ply Stacking Sequence</th>
</tr>
</thead>
<tbody>
<tr>
<td>25/50/25</td>
<td>Quasi Isotropic</td>
<td>[45/0/-45/90]ns</td>
</tr>
<tr>
<td>10/80/10</td>
<td>SOFT</td>
<td>[45/-45/90/45/-45/45/-45/0/-45/45]ns</td>
</tr>
<tr>
<td>40/20/40</td>
<td>HARD</td>
<td>[0/90/0/90/45/-45/90/0/90/0]n</td>
</tr>
</tbody>
</table>

Table 3-4: Summary of Test Results for Laminate Data
**LAMINA and LAMINATE Design Allowables**

- From Lamina Tests the mechanical properties available are:

  - $F_{1tu}$
  - $F_{1cu}$
  - $F_{2tu}$
  - $F_{2cu}$
  - $E_{1t}$
  - $E_{1c}$
  - $E_{2t}$
  - $E_{2c}$
  - $F_{12} \leq 5\%$
  - $F_{12} \leq 0.2\%$
  - $G_{12}$

- From Laminate Tests the mechanical properties available are:

  - OHT (open hole tension)
  - OHC (open hole compression)
  - UNT (Unnotched tension)
  - UNC (Unnotched compression)
  - FHT (filled hole tension)
  - FHC (filled hole compression)
  - Pin Bearing
  - LSBS (Laminate Short Beam Strength)
  - CAI (Compression after impact)
  - ILT (Interlaminate Tension Strength)
  - CBS (Curved Beam Strength)
Replicating Laminate Results

• Using the NCAMP Lamina data as input to a composite analysis program, the NCAMP Laminate layup was recreated.

• This is to show correlation between the composite analysis program and NCAMP Laminate results.

• By obtaining the Laminate Strength Ratio, correlation was shown, therefore other Laminate layups could be used with confidence in the results.
As NCAMP have carried out the full qualification test program, manufacturers are only required to show equivalency.

For each failure type an equivalency test program requires:

1 batch x 8 samples

Equivalency test programs detailed in

**DOT/FAA/AR-03/19 CMH-17**
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Initial Static Load definition

As a fully aerobatic aircraft Swift-VLA is required to withstand +/- 6G Limit Loads.

For initial airframe stressing, this is an ULTIMATE static load of +/-9G.

Initial Fuselage mass estimation established.

Both LIMIT Shear Force and Bending moment diagrams are obtained and initial sizing of the skins and frames could begin.
Initial Skin sizing for the Fuselage

Various sections are taken along the length of the fuselage and I values are established for each section.

Using the initial Bending Moment (M) for each of these X stations, the applied Bending stress ($\sigma_b$) around the fuselage skin is calculated using

$$\sigma_b = \frac{My}{I}$$

From this, the initial skin sizing for the full fuselage can be determined.

Shear is applied over vertical fuselage side only.
Initial Frame Sizing

Simple models using the fuselage shear force diagrams and the geometry of the fuselage at this section were used to make initial sizing of the fuselage frames.

CBAR elements are used to model the frames.

The shear load is evenly distributed down both side of the frames/skin.
Initial Buckling Calculations

With frame positions determined by attachment points of the wing spar, seats, main undercarriage and BRS then the panel lengths for the fuselage skin buckling analysis could be established.

For the initial buckling analysis of the fuselage skin, the fuselage is modelled as two flat sandwich panels (side and bottom sections) with a connecting curved sandwich panel as shown. This determines the panel width. Using a datasheet for Buckling of Rectangular Specially Orthotropic Plates NACA TN 2601 – Compressive Buckling of Simply Supported Curved Plates and Cylinders of Sandwich Construction. The panels are subjected to combined Peak Bending and Shear Stresses calculated previously.

From this analysis a buckling reserve factor could be established. No Buckling up to ultimate.

Local instability calculations are also performed. (detailed later)
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After initial sizing for the fuselage skin and frames, a GFEM was produced using LAMINATE CQUAD elements for the skin and CBAR elements for the frames. FEMAP is used as the pre & post processor.

Density of the glass and foam
For concentrated mass, mass elements are connected to the fuselage using RBEs to replicate the loading expected in the fuselage when at MAUW. e.g. engine, engine frame, instruments, pilots, controls, battery etc

With the GFEM constrained at the wing pick ups, a static load case of +/-9G (Normal G only) is applied to verify the initial sizing of frames and skins.

Solver - MSc Nastran Linear elastic sol101 is utilised
BAR stresses and Laminate Strength Ratios

The GFEM Laminate Strength Ratios and BAR stresses are used to verify the initial sizing.

Failed elements (Strength Ratios<1) identified the areas of the fuselage that will require strengthening, such as at the engine frame attachments to the firewall.
GFEM is used for buckling analysis to identify general instability of the fuselage skin.

MSc Nastran Linear elastic buckling sol105 is used.

With all eigenvalues returned $> 1$, the fuselage skin can be said not to buckle up to ULTIMATE loads.

- The first mid panel buckling result has an eigenvalue = 2.28.
In addition to the buckling analysis carried out by the GFEM, local instability failure modes are considered.

The failure modes that are analysed are:

- Shear Crimping
- Skin Wrinkling
- Core Shear
To allow for more detailed results in regards to the Laminate Strength Ratio analysis and buckling analysis, the GFEM was refined:

- FRAMES C-section meshed as Laminate CQUADs
- Highly Loaded areas Strengthened with thicker layups, around wing pick ups and Main Landing Gear Attachments
- Detailed support brackets added at engine frame attachment points
- BRS parachute support frame added
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Determination of Critical Load cases

Along with an aerobatic flight envelope, the Critical Load cases are determined from CS VLA 423 Manoeuvring Loads and CS-VLA 425 Gust Loads. 16 load cases which 6 are critical envelope cases (SF & BM).

Undercarriage loads have been considered using CS VLA 471 – 499. 6 load cases

BRS parachute loads have been applied to the fuselage as a separate load case.

This resulted in a total of 13 load cases which are applied to the GFEM.
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The loading applied during checked manoeuvres include:

- Angular Accelerations
- Normal Accelerations
- Wing Lift
- Horizontal Tail Load
- Zero Lift Moment
- Rear Spar Pick Up Balancing Load

The GFEM has a single point constraint at the CG which is attached to the GFEM using RBEs.

For a balanced load case this results in zero SPC forces at the CG.
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Analysing Results

For each critical load case the following is extracted from the GFEM:

- Fuselage Deflections
- Laminate MIN Strength Ratios
- Eigenvalues from Buckling Analysis
Weight saving

After the critical load cases are analysed, the GFEM can be optimised to reduce fuselage weight.

One method used to achieve this was through the reduction in foam thickness for the FWD fuselage skin sections.
Thanks for your attention