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Cessna 441 Life Extension

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Introduction

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Cessna 441 is a twin turboprop, pressurised aircraft.

In 2007, as a result of a Cessna Supplemental Inspection Document (SID), all 441s in Australia with over 22,500 hours flight time were grounded by CASA.

This life is now also included in the Cessna system of maintenance.

Several aircraft were grounded immediately, and others are close to 22,500 hrs

CASA Instrument

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Direction — Cessna 441 Conquest

1 Commencement

This instrument commences on 25 August 2007.

2 Application

This instrument applies to each operator, and each pilot, of a Cessna 441 Conquest (the ***aircraft***) that:

(a) has 22 500 hours or more time-in-service; and

(b) is:

(i) an Australian aircraft; or

(ii) a foreign registered aircraft in Australian territory.

3 Direction

The aircraft may not be flown at any time or for any purpose while this direction is in force unless, being satisfied that it is safe to do so, the Director approves in writing the details of the flight, including the time, route, purpose and risk mitigators.

Note 1 The aircraft manufacturer, Cessna Aircraft Corporation, has issued a Supplemental Inspection Document (SID) which recommends that the aircraft be retired when it has accumulated 22 500 flight hours because continued airworthiness can no longer be assured.

Background

The Cessna Aircraft Corporation (the ***manufacturer***) has recently issued a Supplemental Inspection Document (***SID***) for the Cessna 441 Conquest which recommends that the aircraft be retired when it has accumulated 22 500 flight hours because continued airworthiness can no longer be assured due to the aircraft's structural limitations at this level of usage. The manufacturer has not developed a remedial maintenance program.

CASA considers that, in view of the manufacturer's recommendation, these aircraft should be immediately grounded in the interests of the safety of air navigation.

This talk will cover:

- The basis of the life extension program
- STC holder responsibilities
- Methods used to develop the program
- Critical structural areas
- Finite element modelling
- Wing strain gauge testing
- Fuselage frame modifications
- Fatigue and damage tolerance calculation methods
- Pressure bulkhead reinforcing
- Widespread fatigue damage considerations
- Fuselage stringer considerations

Life extension program

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TAE is an organisation which, among many other activities, maintains Cessna 441s. It was previously Tenix Aviation.

TAE and AEA are cooperating to develop a life extension program for the 441

The **methods** of life extension include:

- Inspections
- Replacement of components with standard or redesigned new parts
- Reinforcement of existing components

Outcome of the program will be a Supplemental Type Certificate (STC), which will cover:

- Physical modification of aircraft
- New IFCA incorporating additional inspections

Aims of the life extension program include:

- Increase life from 22,500 hours to 40,000 hours/cycles
- Make the structure damage tolerant where practical

Life extension program

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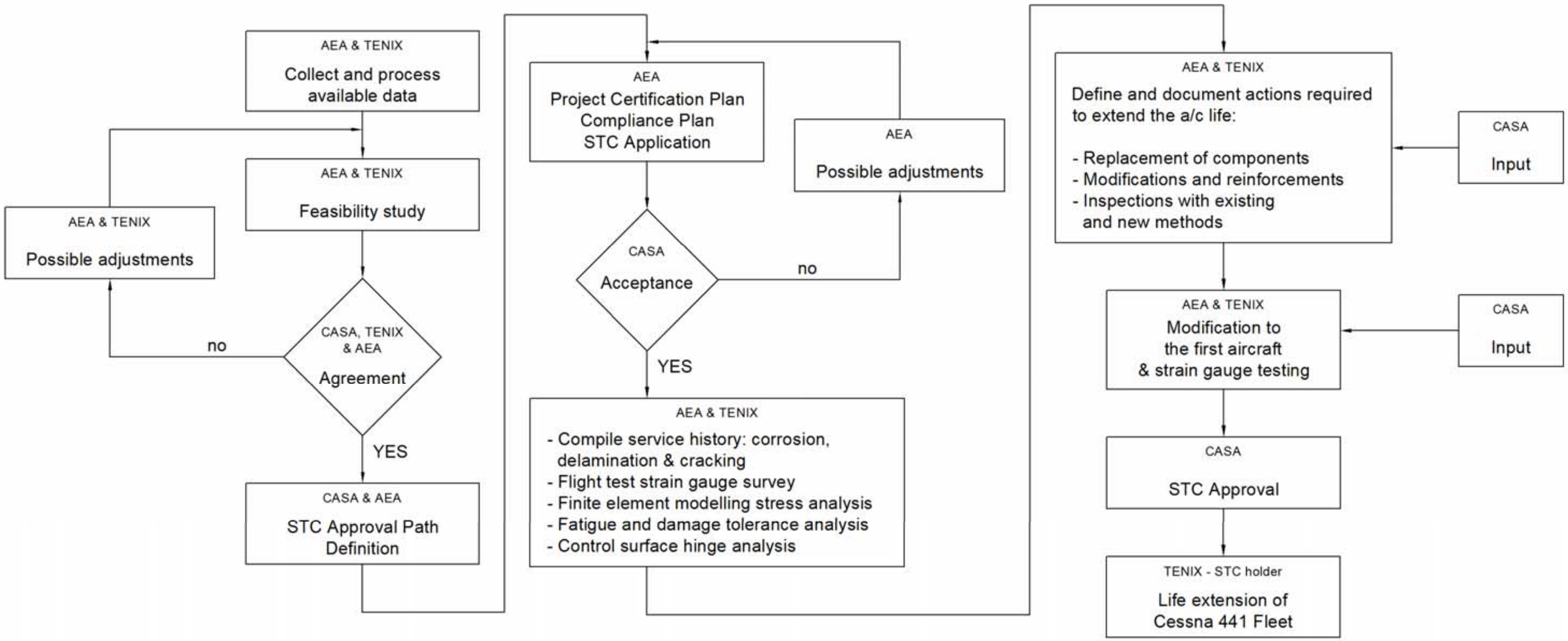
The process for the life extension, involving AEA, TAE and CASA, is:



What are the STC holder's responsibilities?

Methods

A major problem with the life extension program is that is has been done without support from the OEM.

The general approach to the program was:

1. Establish reason for life limitation
   - Structure
     - Fatigue, Corrosion
   - Systems
     - Electrical, hydraulic etc
2. Find problem areas
3. Choose life extension methods for each area identified
4. Show that the method is effective

NOTE: It is more than just a structural evaluation.
Cover the entire aircraft.
Focus on maintainability.

What are the problem areas?

There are several sources of data which indicate the locations and nature of problem areas, both structural and other.

Cessna data

- Maintenance Manual
- Service Bulletins
- Supplemental Inspection Documents (SIDs)

Local service experience

- TAE specialises in maintenance of Cessna 441s, and has gained wide knowledge of recurrent problems
- AEA has done a large number of repairs to 441s over many years, and has a large database of repairs
- All systems considered

Structural analysis and test.

- Finite element analysis
- Strain gauge testing

What are not problem areas?

In addition to locations which may have problems, there is evidence which shows that an area will not have problems at the extended life.

This is more applicable to systems rather than structure.

In particular, the Cessna 404 is very similar to the 441, and has a life of 40,000 hours. Many systems on the two aircraft are the same.

The primary differences between the two aircraft are that the 404 has piston engines rather than turboprops, and is not pressurised.

The wings have similar structure, but the 441 has increased span, higher gross weight, greater fuel weight, lighter engines and different flight envelope. A comparison of the wing loads on the two aircraft was done, but the differences meant that there was no relationship between the fatigue lives.

Life extension methods

Inspection

- Continue existing inspections
- New inspections
- Damage tolerance principles in redesigned parts enhances inspectability

Part replacement

- Standard new parts. The new part will have a life of 22,500 hours

Modifications

- Replace sections. Some parts, such as fuselage frames, cannot be removed and replaced in one piece. It is possible to cut out the critical section of the part, and replace with a new section. The replacement section can be equivalent to or the same as the original part, or may be stronger
- Add reinforcement to reduce stress.

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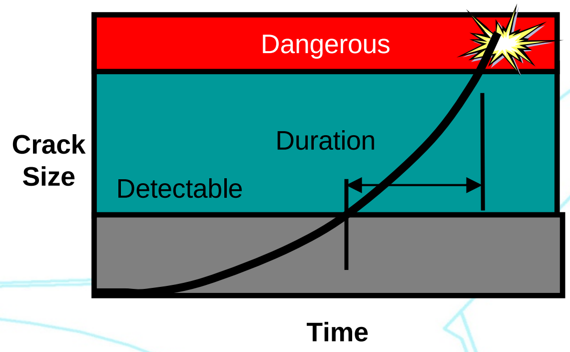
Structure fatigue - certification history

The rules in FAR 23 relating to fatigue in Normal category aircraft have developed progressively. The main changes were:

- From the initial issue of FAR 23 in 1964, pressurised cabins were required to be safe life or fail safe (23.571).
- In amendment 7 in 1969, this requirement was extended to include wing, carrythrough and attaching structure (23.572).
- Amendment 14 in 1973, required test justification for safe life structure except for simple structures.
- Amendment 38 in 1989 added empennage.
- In amendment 45 in 1993, the option of damage tolerance was added

Although still acceptable, fail safe design by itself is generally regarded as unsatisfactory, since it does not give a rational method of finding a failure.

Structure fatigue - Certification



Damage tolerance adds the requirement that damage should be found before it becomes dangerous.

The certification basis of the Cessna 441 is FAR 23 Amendment 14.

The fuselage and wing were designed and tested to be fail safe.

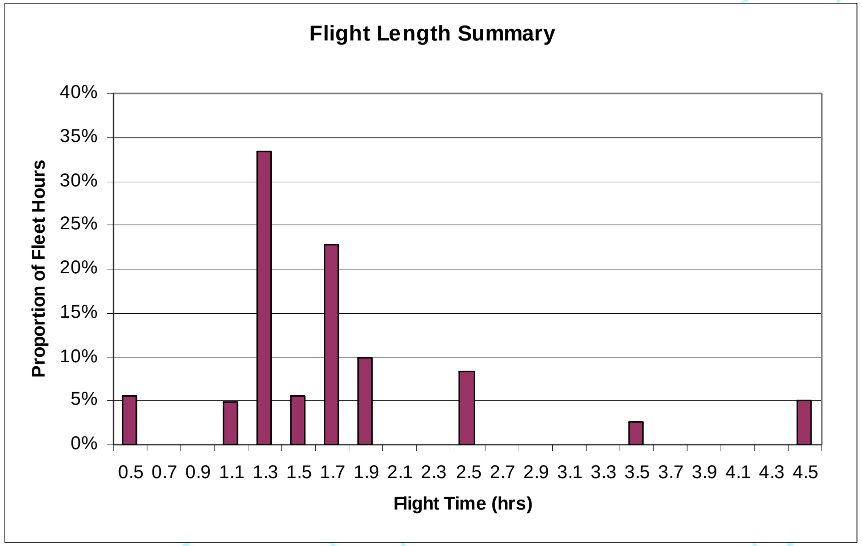
In particular, the wing design was a great improvement on the Cessna 402 wing, which has a single spar, is a safe life structure, and has a short life.

Structural analysis - load spectrum

An important part of the fatigue analysis is development of load spectra.

Applied loads are:

- Flight loads
  - Gust
  - Manoeuvre
- Pressurisation
- Ground loads
  - Landing
  - Taxi
- Ground - Air - Ground (GAG)



Spectra were derived from:

- AC23-13A *Fatigue, Fail-Safe, and Damage Tolerance Evaluation of Metallic Structure for Normal, Utility, Acrobatic, and Commuter Category Airplanes*
- Operator surveys

For most fuselage structure, pressurisation loads are dominant.

Fatigue spectrum

AC23-13A method, which uses one stress value for whole flight, was modified to account for different loadings.

- Max zero fuel weight
- Maximum fuel
- Half way between

Weighted mean life proportioning the 3 loadings based on operator survey

Wing Loads

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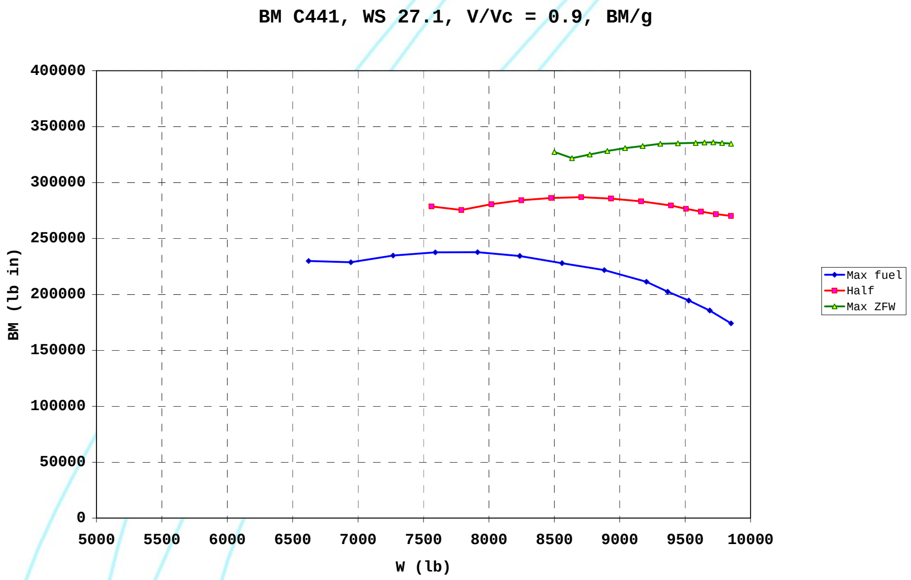
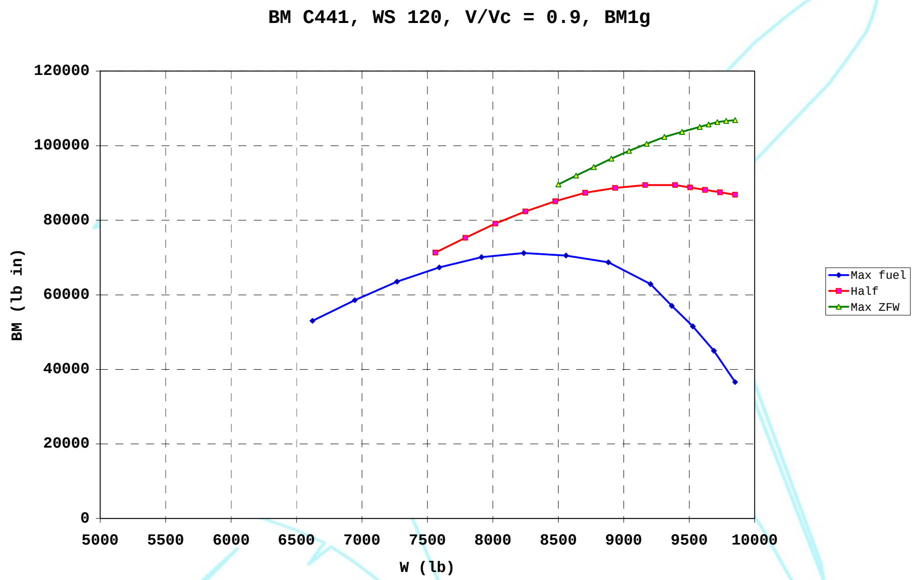
Wing loads and stress depends on Weight W and Zero Fuel Weight ZFW





Bending moment at two wing stations is plotted against aircraft weight W, for different initial fuel weights.

Each curve corresponds to burning fuel from MTOW down to ZFW.

Maximum loads can occur at intermediate fuel loadings.

Operator Flight Survey

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From AC23-13A, flight time is assumed to be 1.1 hours, for all fuel loadings.

Shorter flights increase the fatigue damage which is caused by the GAG cycle, so that shorter flights give lower lives.

A flight with high fuel loading may be either a single long flight, or a series of shorter flights without refuelling.

Finite Element Models

Structural analysis was done using several finite element models

These models were

- **Coarse mesh model** of whole airframe. For convenience, this was divided into two parts
  - Fuselage and wing
  - Tailplane and fin
- **Detail fine mesh models**
  - Fuselage frame and skin
  - Forward pressure bulkhead top
  - Stringer

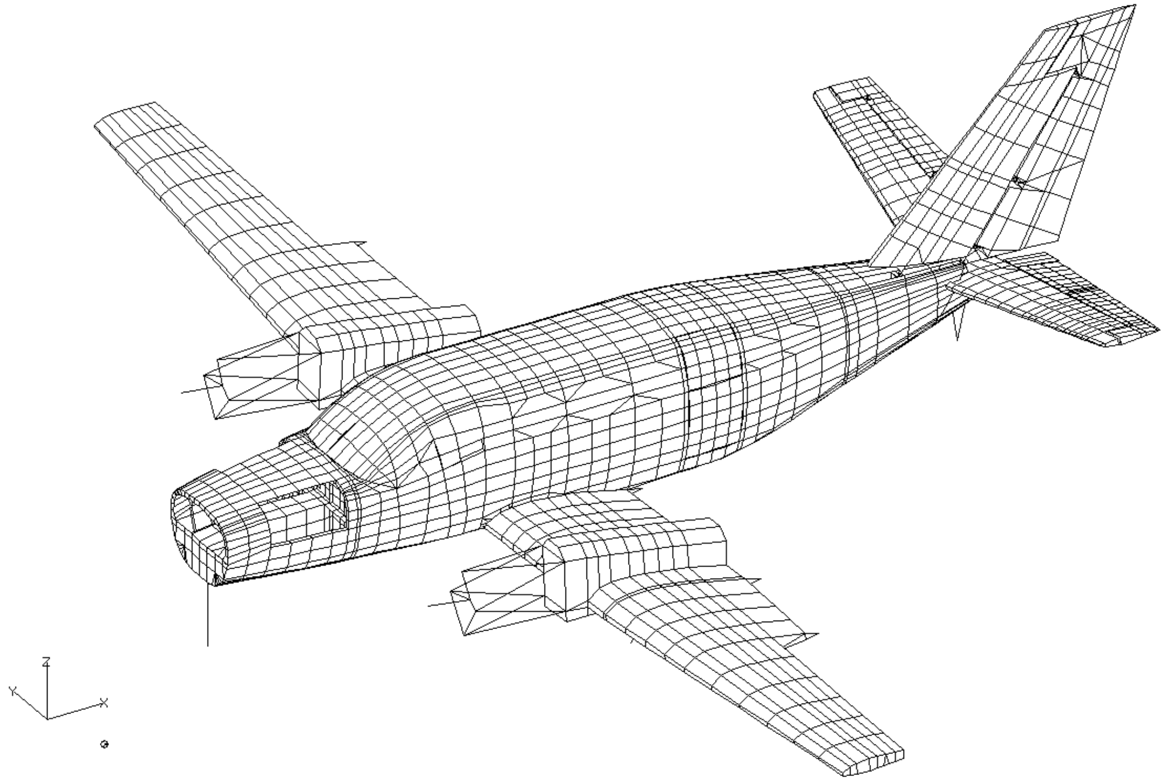
The detail models are described later.

FEM - Main Model

The main model is a coarse mesh model of whole airframe.

The model is run in two configurations:

- Standard aircraft
- Modified aircraft



FEM - Main Model

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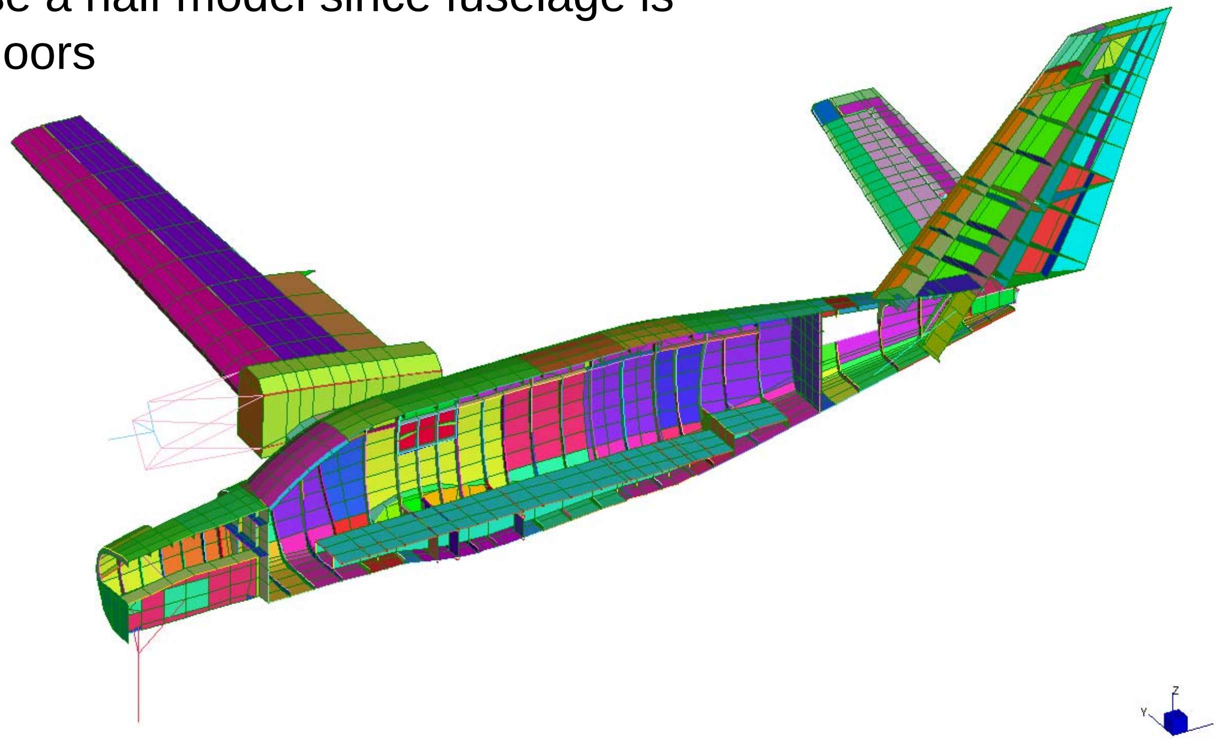
Model follows aircraft structure

- Fuselage frames, stringers
- Wing and tail ribs, spars, stringers
- It is not possible to use a half model since fuselage is unsymmetric due to doors

Size of model:

- 6115 nodes
- 15496 elements
- 1194 properties



A large part of the modelling time was spent calculating geometry and properties

Data is mainly from measurements of the aircraft

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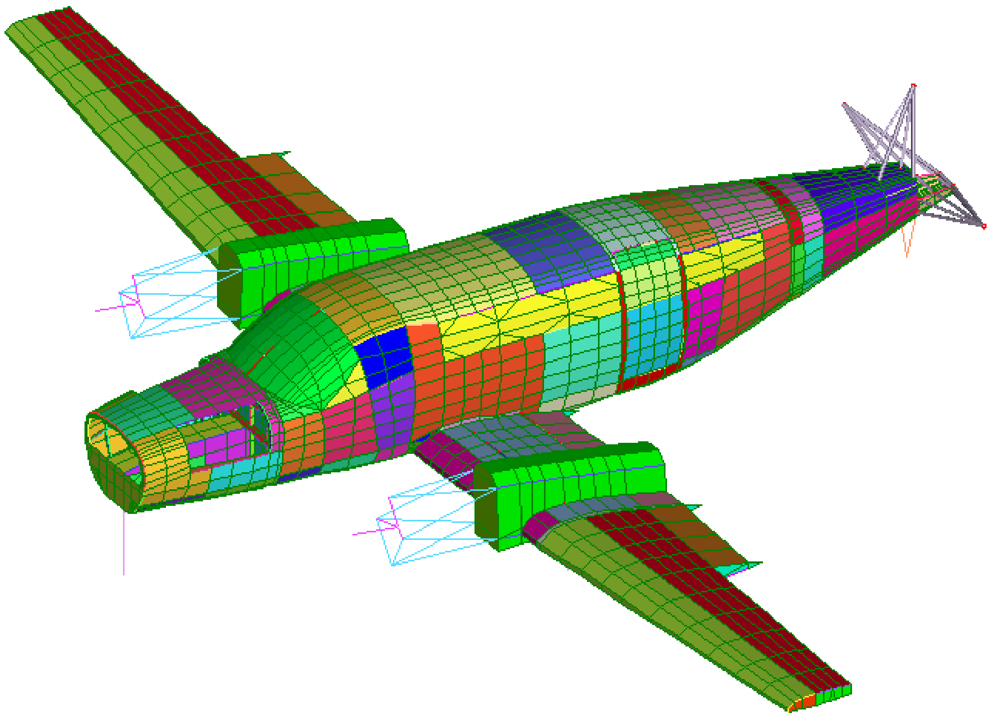
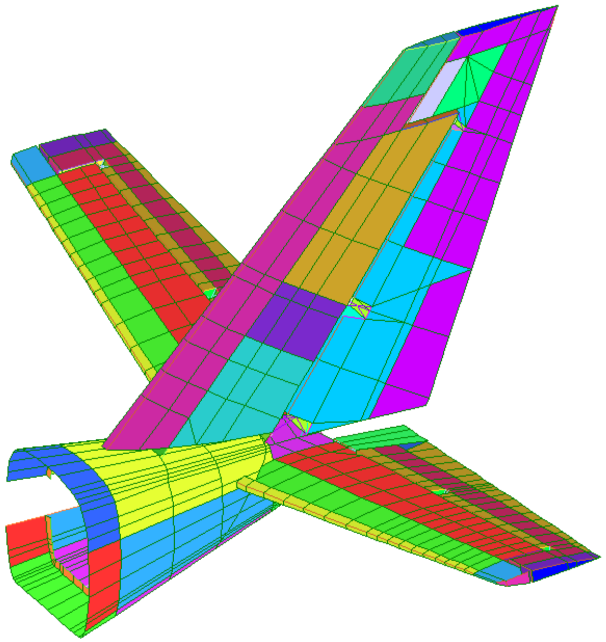
FEM - Main Model

For convenience, the analysis was divided into two parts

- Fuselage and wing. Tail loads were applied through a simple beam representation of the tailplane and fin.
- Tail. The tailplane and fin models were attached to the rear fuselage.





FEM - Main Model Constraints

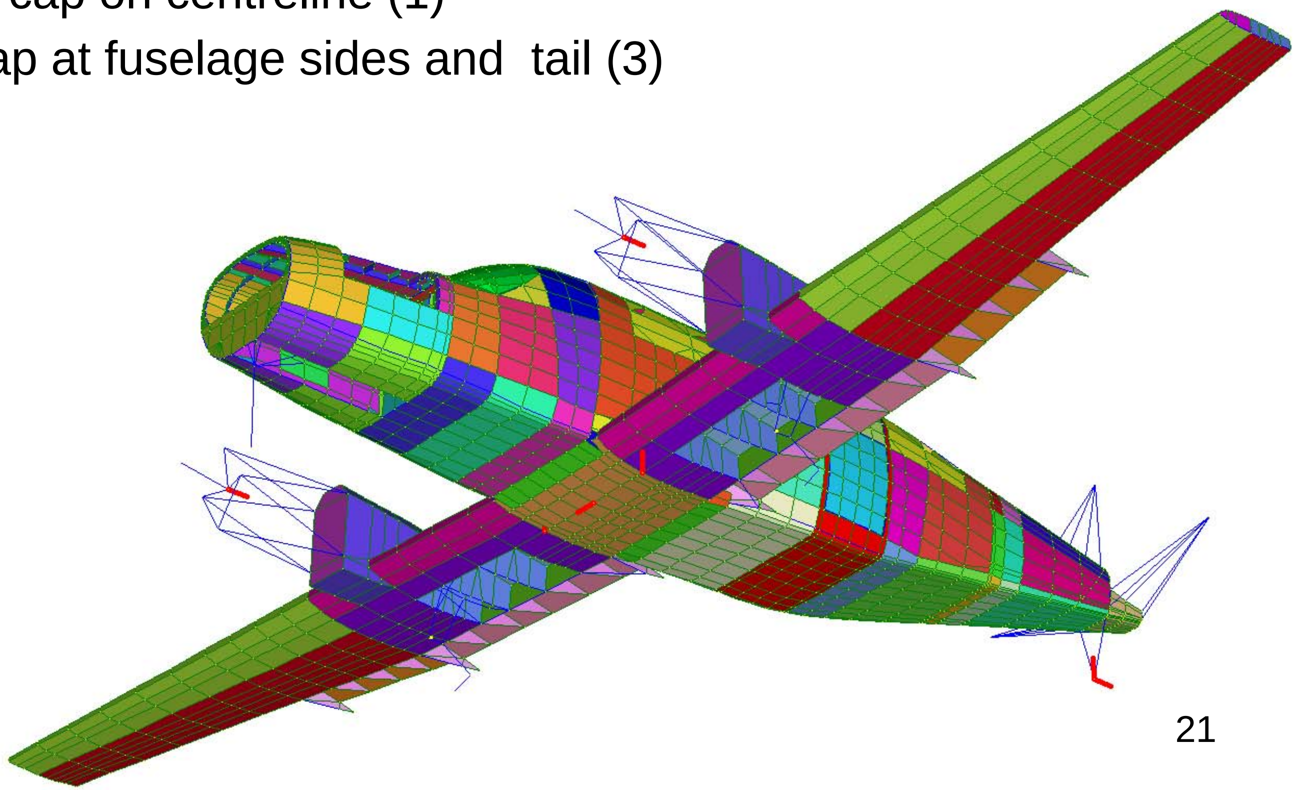
Loads were applied as balanced load cases.

Supports were statically determinate.

Reactions were zero except for small errors.

Constraint directions

- x Engines (2)
- y Bottom spar cap on centreline (1)
- z Main spar cap at fuselage sides and tail (3)



FEM - Main Model Unit Load cases

Unit load cases were run in the model, and combined for particular loading and flight conditions

Unit load cases are:

ID	Description		ID	Description
1	1g inertia - empty aircraft		26	Wing chordwise load flaps 20deg (Px)
2	Payload 1		27	Wing additional lift- flaps 30deg (Pz)
3	Payload 2		28	Wing basic lift - flaps 30deg (Pz)
4	Payload 3		29	Wing chordwise load flaps 30deg (Px)
5	Payload 4		31	Fuselage pitching moment
9	X direction inertia loads		32	Fuselage drag
11	X direction inertia loads - Fuel		35	Engine thrust
15	Fuel 20%		36	Engine torque
16	Fuel 40%		41	Left Tailplane load
17	Fuel 60%		42	Right Tailplane load
18	Fuel 80%		43	Fin load
19	Fuel 100%		61	Main and nose undercarriage
21	Wing additional lift- flaps up (Pz)		62	Main undercarriage Tail down landing
22	Wing basic lift - flaps up (Pz)		63	Main undercarriage Level landing
23	Wing chordwise load (Px)		64	Main and nose undercarriage
24	Wing additional lift- flaps 20deg (Pz)		99	Pressurisation
25	Wing basic lift - flaps 20deg (Pz)			

FEM - Combined Load cases

Factors for combining load cases were derived from loading and flight conditions

The load cases considered for fatigue and stress spectrum are:

1 g Cruise (includes pressurisaton)
Gust 1g increment
Manoeuvre 1g increment
1 g Flaps extended Vf
1 g Ground (taxi)
10 ft/s landing Type 1 (Tail down)
10 ft/s landing Type 2 (Level landing with drag)

- 1g cruise is different from 1g increment because of stress at zero g, mainly from section pitching moment and wing twist.
- Gusts and manoeuvre loads different since tail loads are different.
- 1g taxi stress loads correspond to static equilibrium on wheels
- For landings, wing lift = 0.67 * W

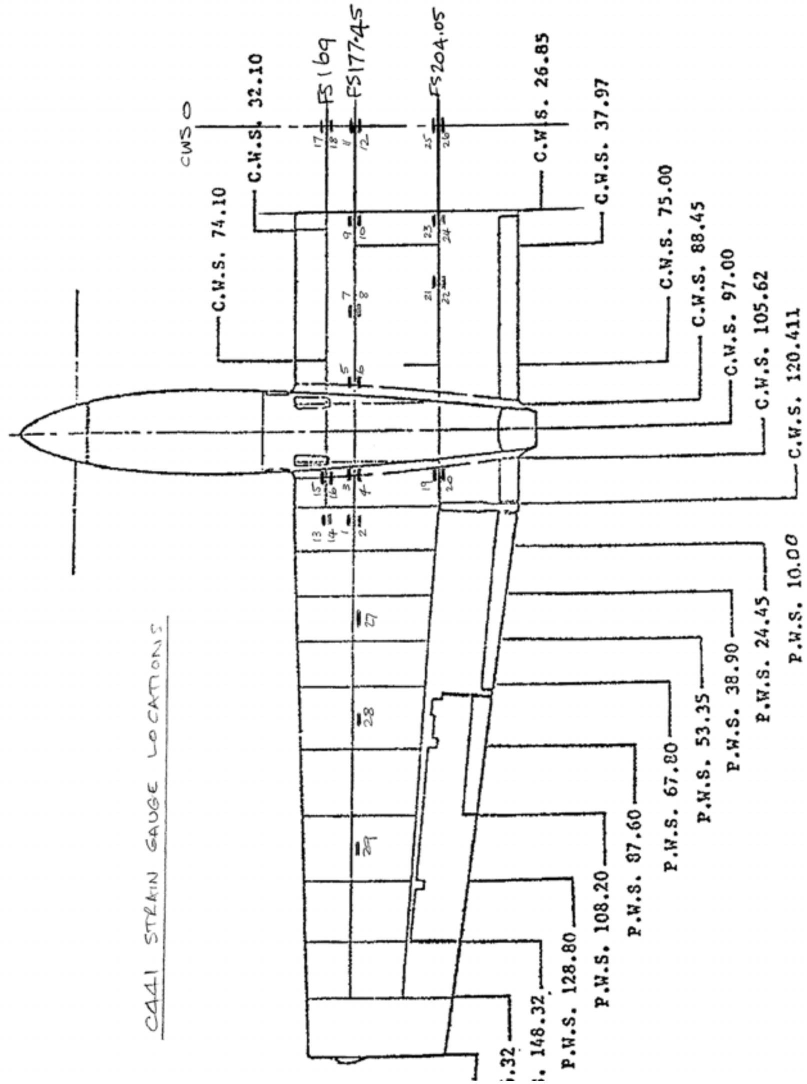
Wing Strain gauge testing

The FEM is validated by comparing results to strain gauge tests.

Two sets of tests were done:

Wing

- Flight test
- Manoeuvres at various weights and fuel load
- Speed
- Flap settings



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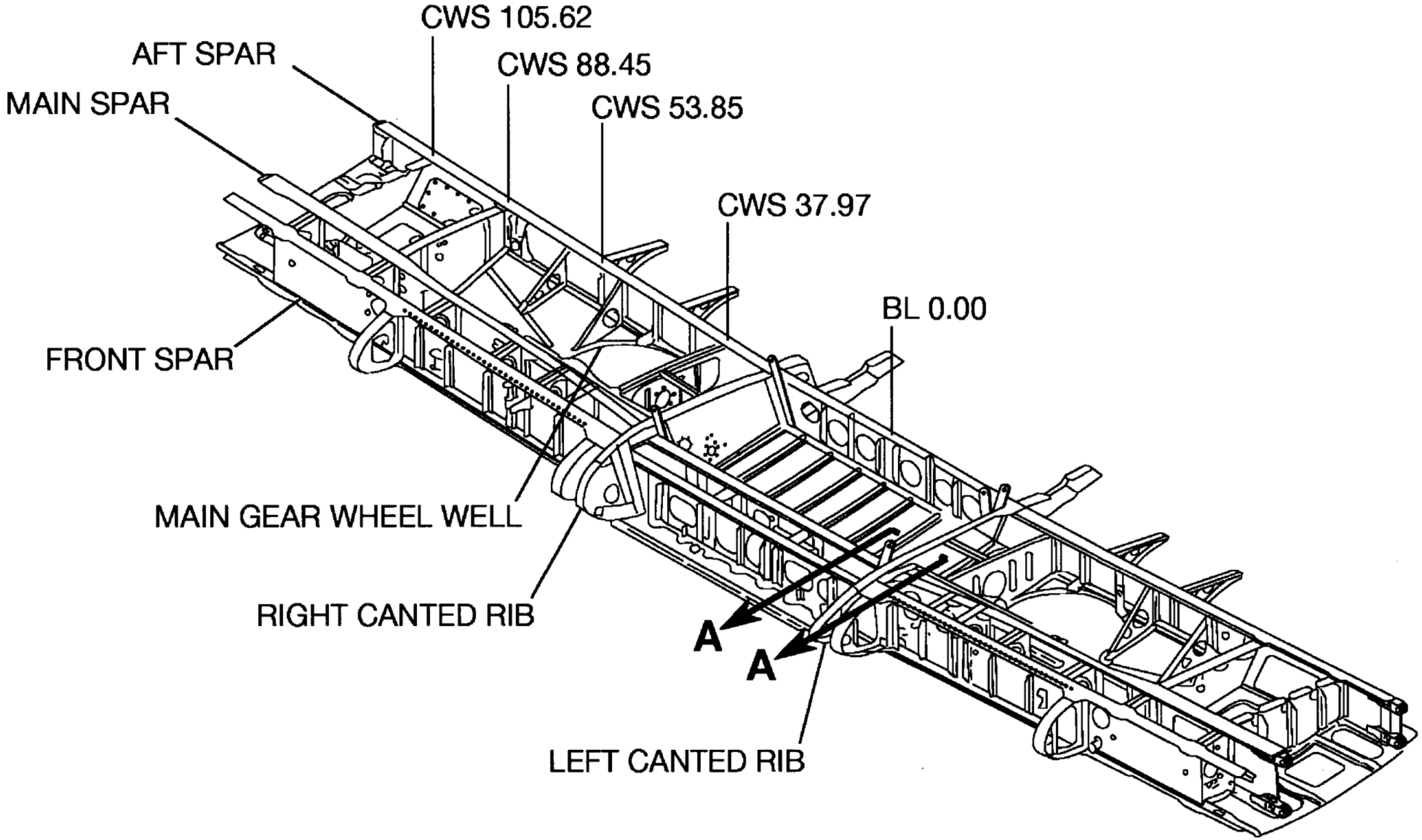
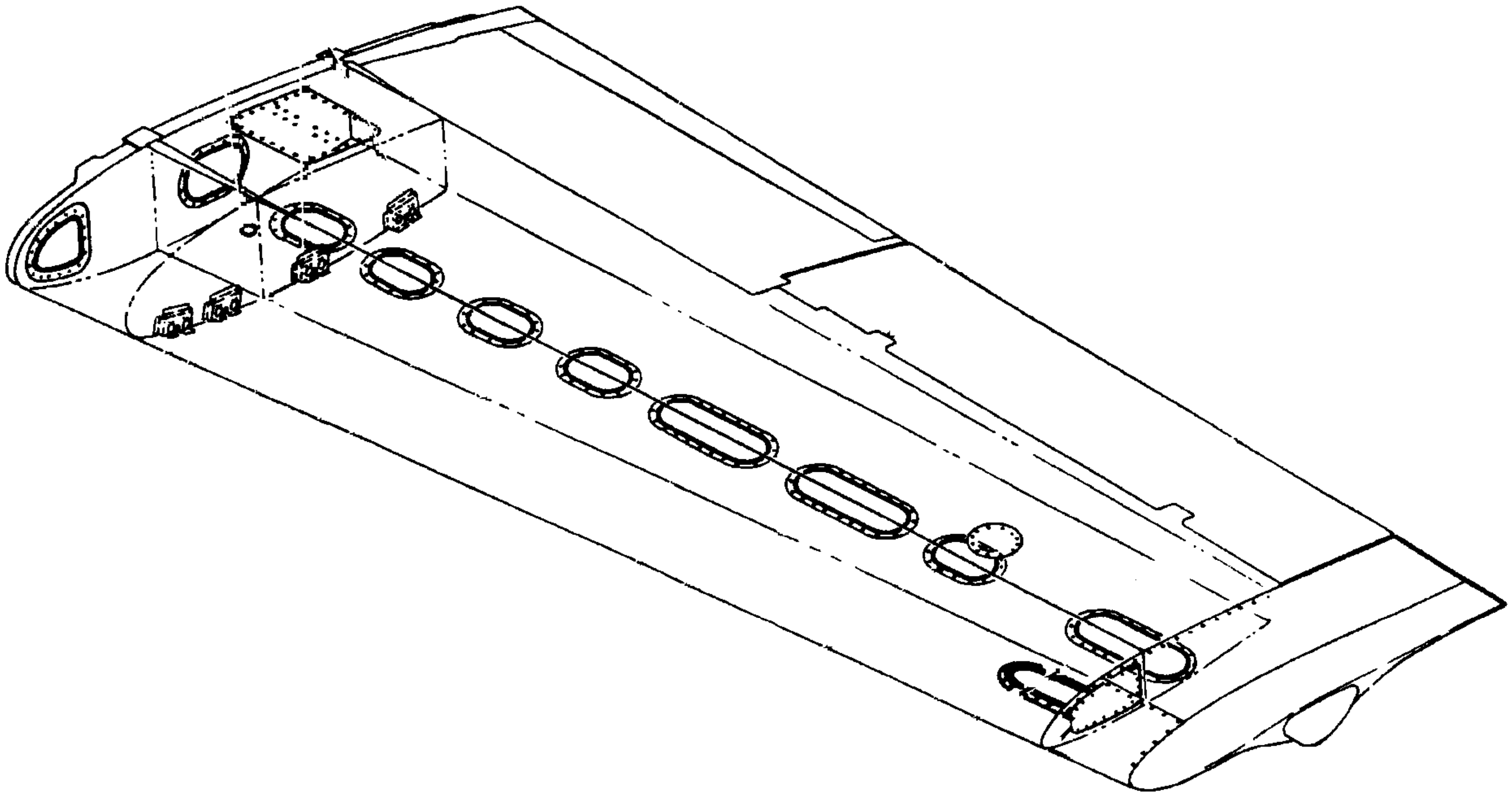
Wing centre section construction



Outer wing construction



Wing strain gauge testing

Strain gauge results are compared to FEM results in order to validate the FEM.

In order to do this,

- Strain gauge results must be processed to get the desired quantities, which are **1g stress** and **stress per g**, for various loadings and flight conditions
- Corresponding load cases must be run in the FEM

Strain gauge results must be processed

- Some gauge results were incorrect
  - Gauge not working - results discarded
  - Results wrong sign = multiply by -1
- Where there are two gauges on front and back flanges of T section, the average stress from them was used
- Results from multiple test points must be combined to give the desired quantities, which are stress per g and 1g stress.

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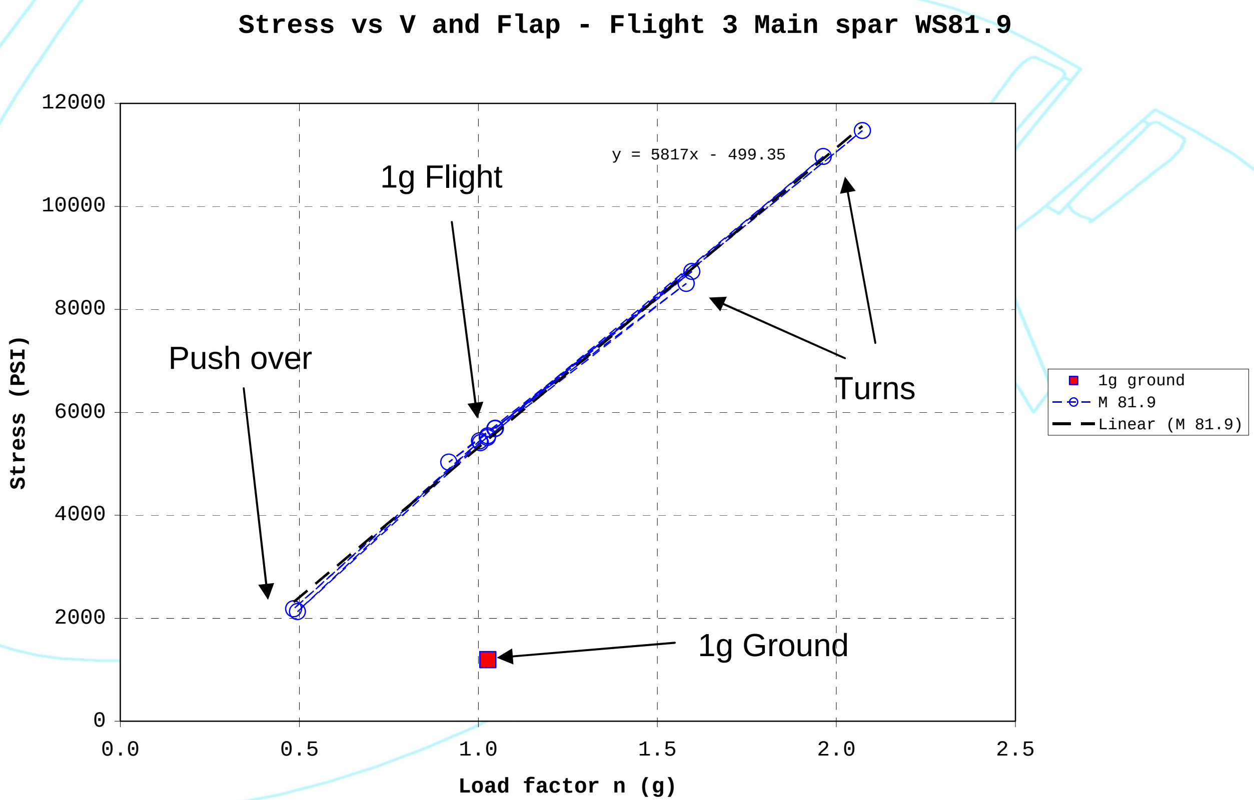
Wing strain gauge processing

Desired quantities are 1g stress and stress per g

In example below, results are reasonably linear. Best fit line is

	S	= 5817 * n - 499	
So	Spg	= 5817 (lb/in2)/g	
	S1g	= 5817 * 1 - 499	= 5318 lb/in2



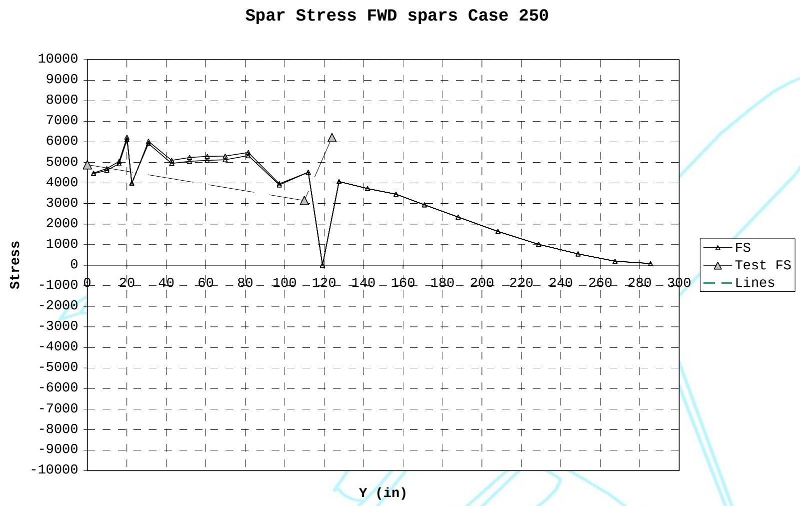
Wing strain gauge correlation

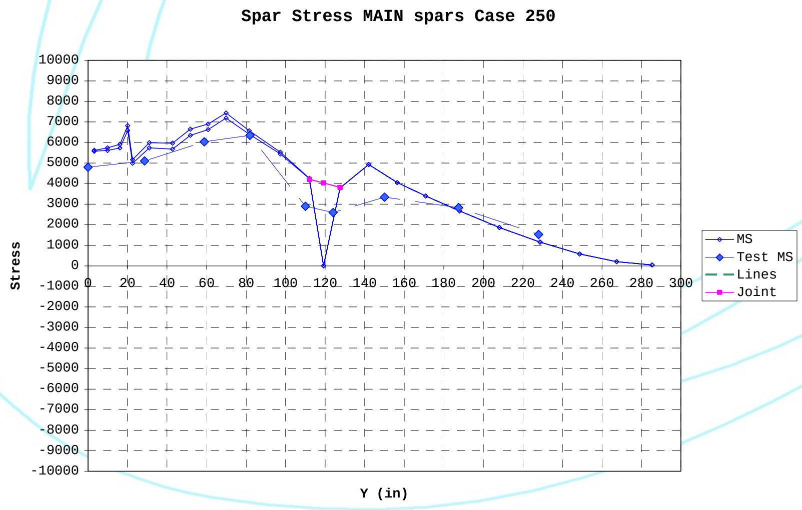
Strain gauge readings for forward, main and rear spars are shown, for one flight test.

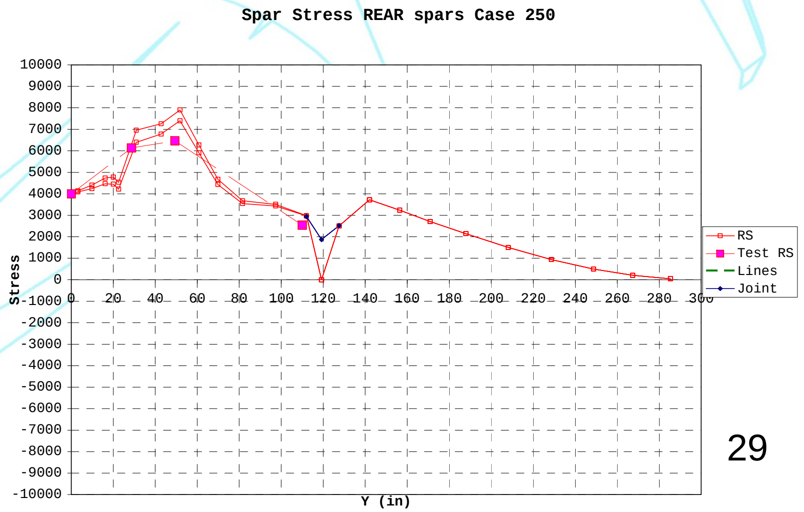
These curves are for **stress per g**.

The zero value at WS120 is a result of the modelling technique used at the outer wing joint.

Agreement is good except for a point on the forward spar just outboard of the joint. This is due to the offset of the stringer at the joint.







Wing strain gauge processing

It is not possible to measure 1g stress accurately

- The aircraft weight is on the undercarriage when the strain gauges are installed, so there is an initial 1g ground stress on the wing.
- Gauges must be zeroed, also with weight on the undercarriage. This would probably be done for each set of measurements.

The incremental stress Spg can be measured accurately, since the zero error does not affect the slope of the curve of stress vs load factor.

If the gauges are zeroed on the ground, then the stress measurement is relative to the 1g ground stress.

For the purpose of correlation, we can compare the increment in stress between ground and flight, ie (S1g flight - S1g ground)

Wing strain gauge correlation

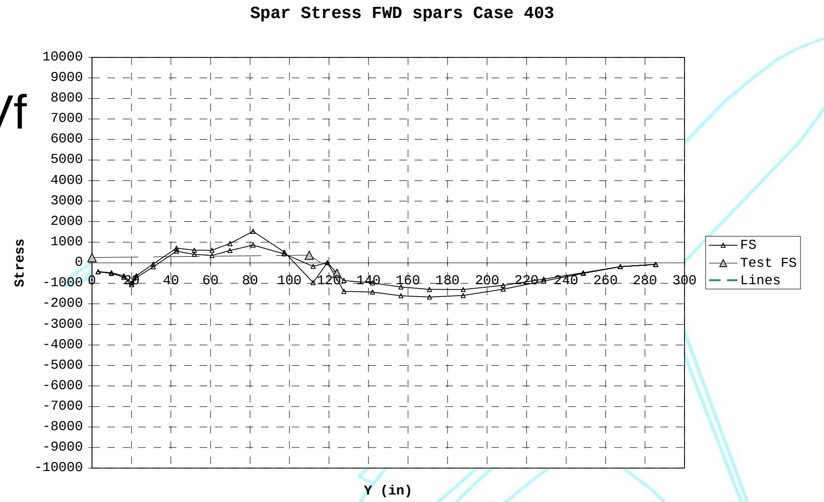
Curves are for

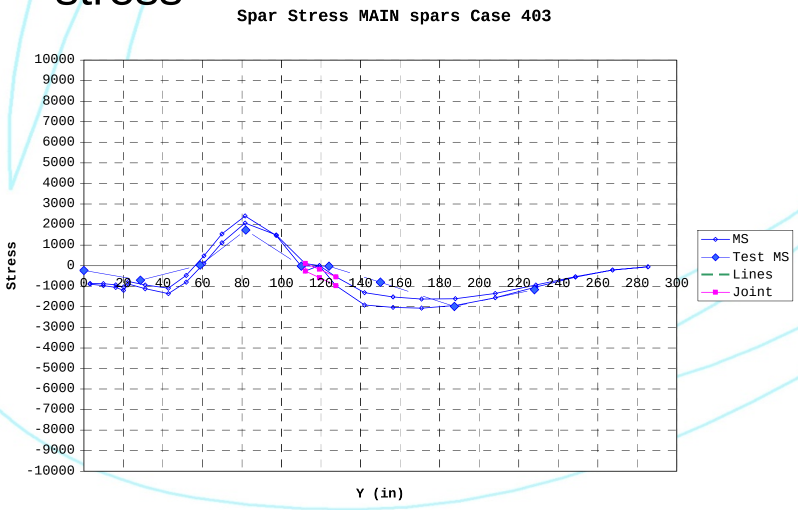
(S1g flight - S1g ground)

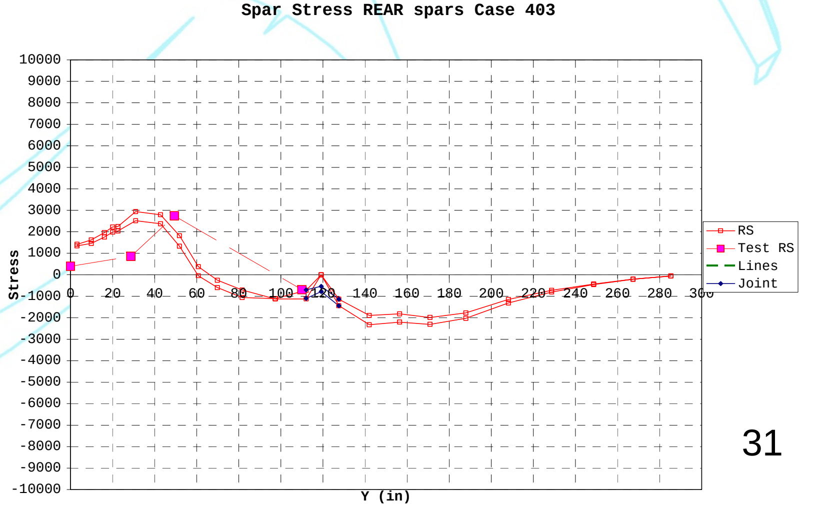
These curves are for **30 degree flap**, at Vf

The effect of flap is:

- Lift is moved inboard. The outer wing panel actually has a down load at 1g.
- High torsion due to flap pitching moment is partly taken by differential bending of the main and rear spars, giving rapid spanwise variation of spar stress







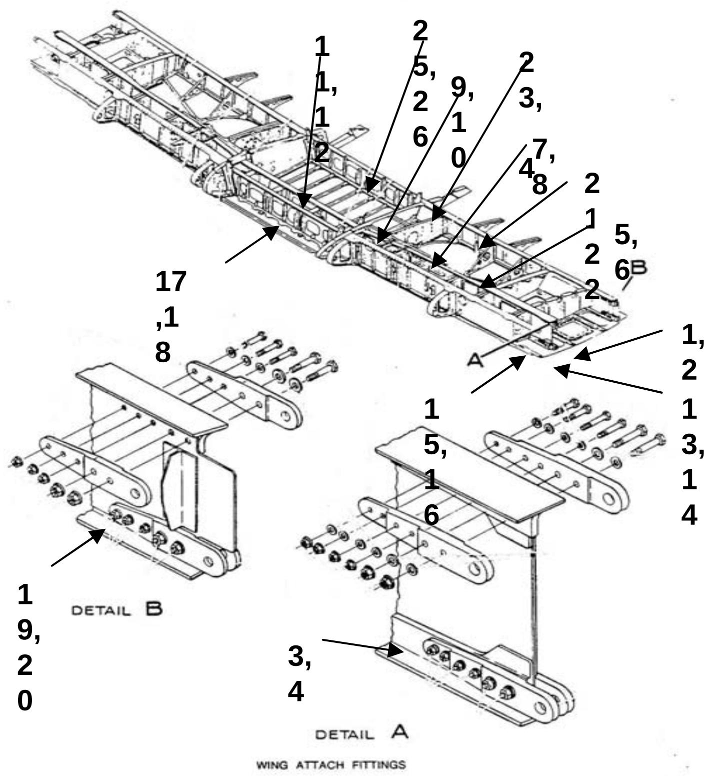
Wing fatigue

Damage tolerance analysis was done on the wing spars and joint fittings.

It was found that the existing Cessna SID inspections were basically adequate to ensure continued safety.

The majority of the wing structure has a safe life of greater than 40,000 hours, in addition to being fail safe.

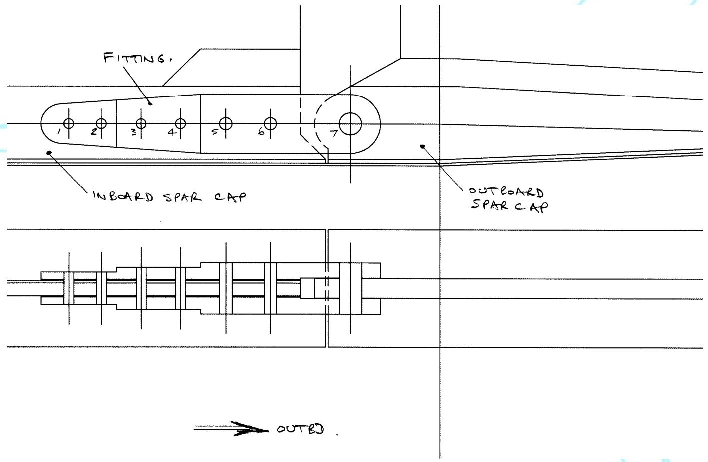
The existing inspection of the wing attach fittings has been extended to additional locations.

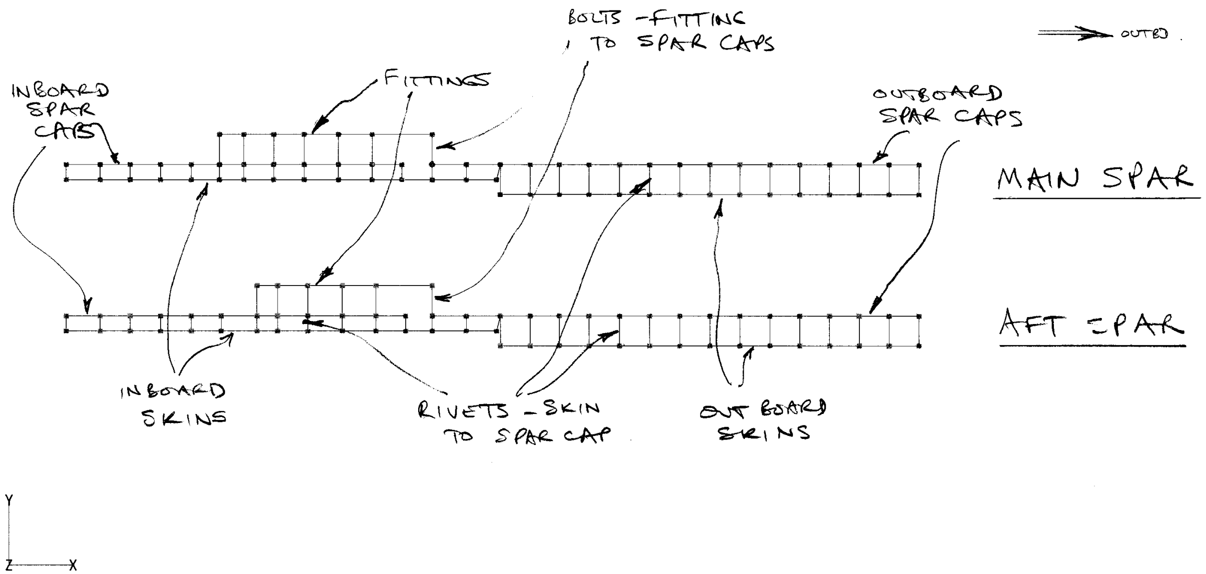


Wing attach fittings

Wing attach fittings were analysed using a simple 1 dimensional FEM, in which fitting axial stiffness and fastener stiffness was simulated. This enabled calculation of bearing and bypass stress at each fastener hole.

These stresses were used for the crack growth analysis of the spar caps and fittings.





Fuselage

The main areas in the fuselage which are fatigue critical are:

- Frames
- Door and window frames
- Forward pressure bulkhead
- Bottom stringers

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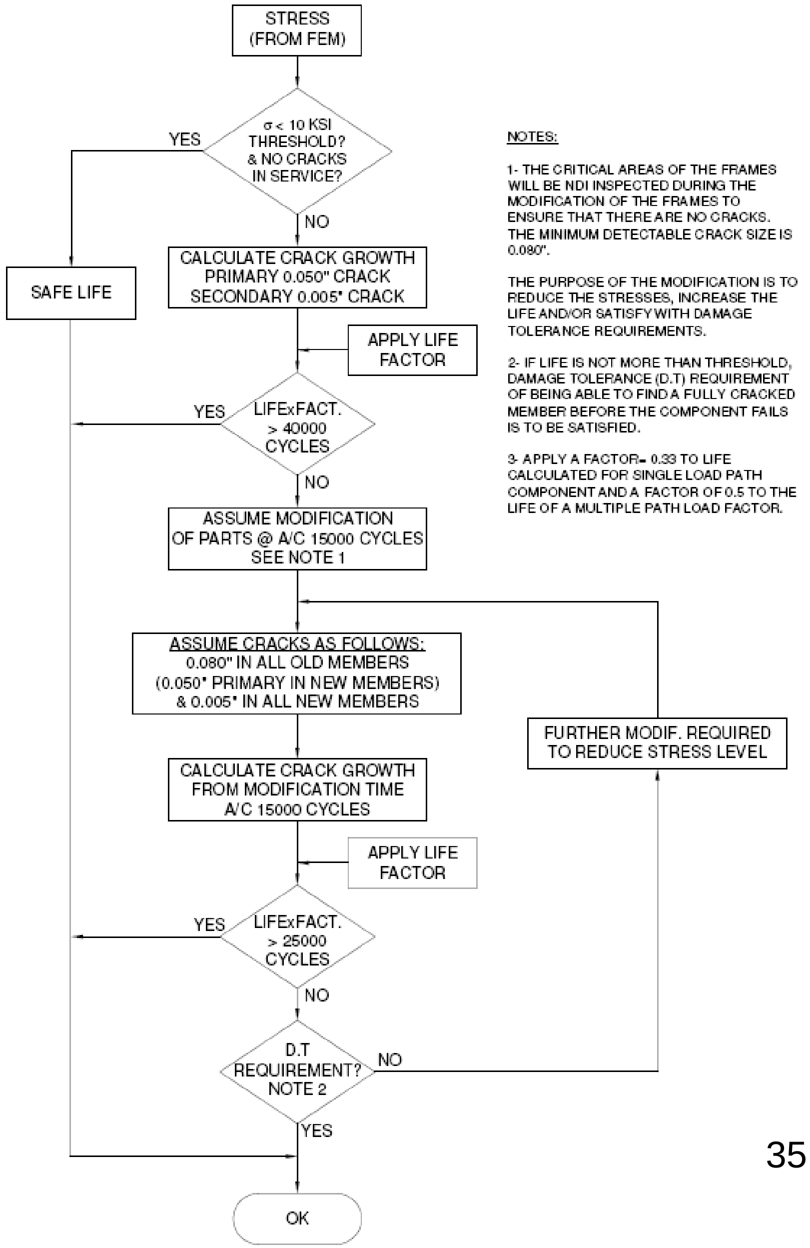
Classification of Fuselage Structure

Structure can classified according to the load spectrum, stress level and structural detail.

The flowchart applies to fuselage structure, primarily loaded by pressurisation.

Calculation of SN life showed that a stress of less than 10 KSI for normal pressure differential, gives a life of greater than 40,000 cycles.

Higher stress, or a history of cracking, required modification and further analysis.



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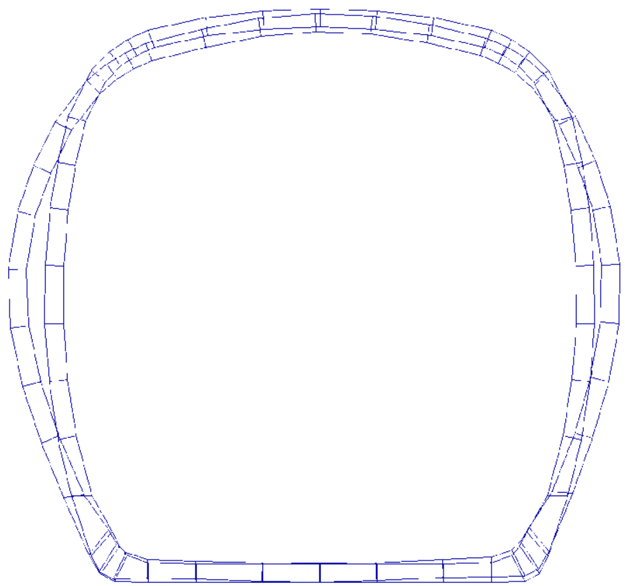
Fuselage frames - stress

Under the effect of the cabin pressure, the fuselage, whose cross section is a rounded rectangle , tends to become circular.

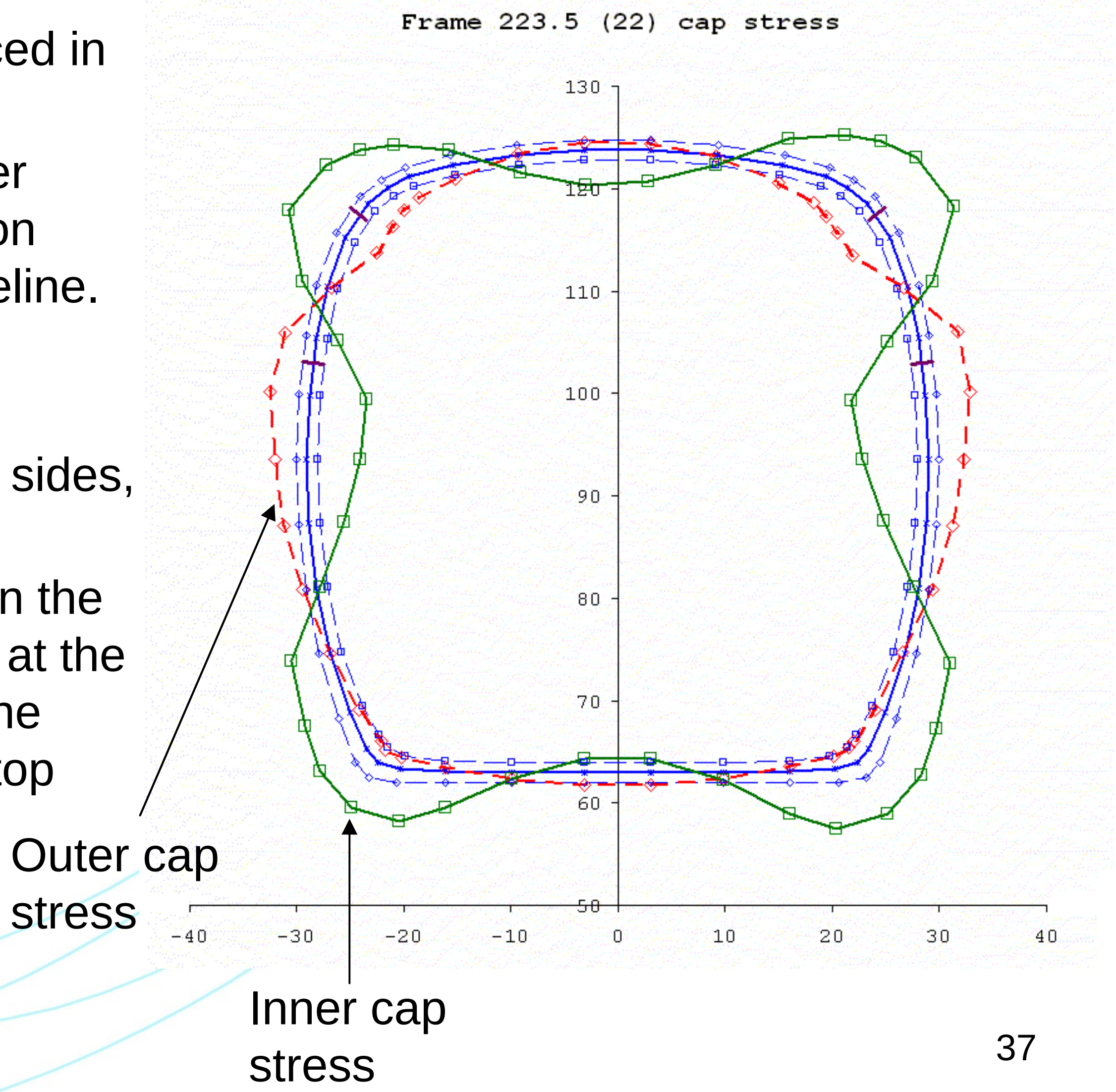
The sides and top and bottom bulge outward, and the corners straighten.



Frame deflected shape

Fuselage frames - stress

- Bending moments are produced in the frame.
- Stresses in the inner and outer caps is plotted, with tension outward from frame centreline.
- Inner cap is in tension at the corners
- Outer cap is in tension on the sides, top and bottom.
- In general, the tensile stress in the inner flange of the frames at the corners is higher than in the outer flange at the sides, top and bottom.

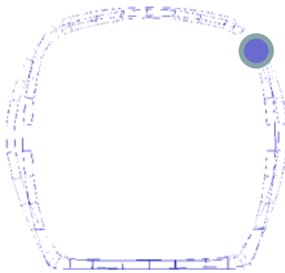


Fuselage frames - failsafe

There are three failsafe mechanisms or behaviours shown by the frames.

1. Multiple local load paths: each frame has more than one member or load path at each of the critical locations.

2. Pin joint / hinge effect: when a flange fails, the other flange and skin or strap (as applicable) work as a pin joint connection, at the failure location. The load redistributes along the frame.



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Fuselage frames - failsafe

3. Load transfer to adjacent frames: frames adjacent to the failed frame will take additional loading.

The failsafe strength of the fuselage was simulated by removing elements in the FEM

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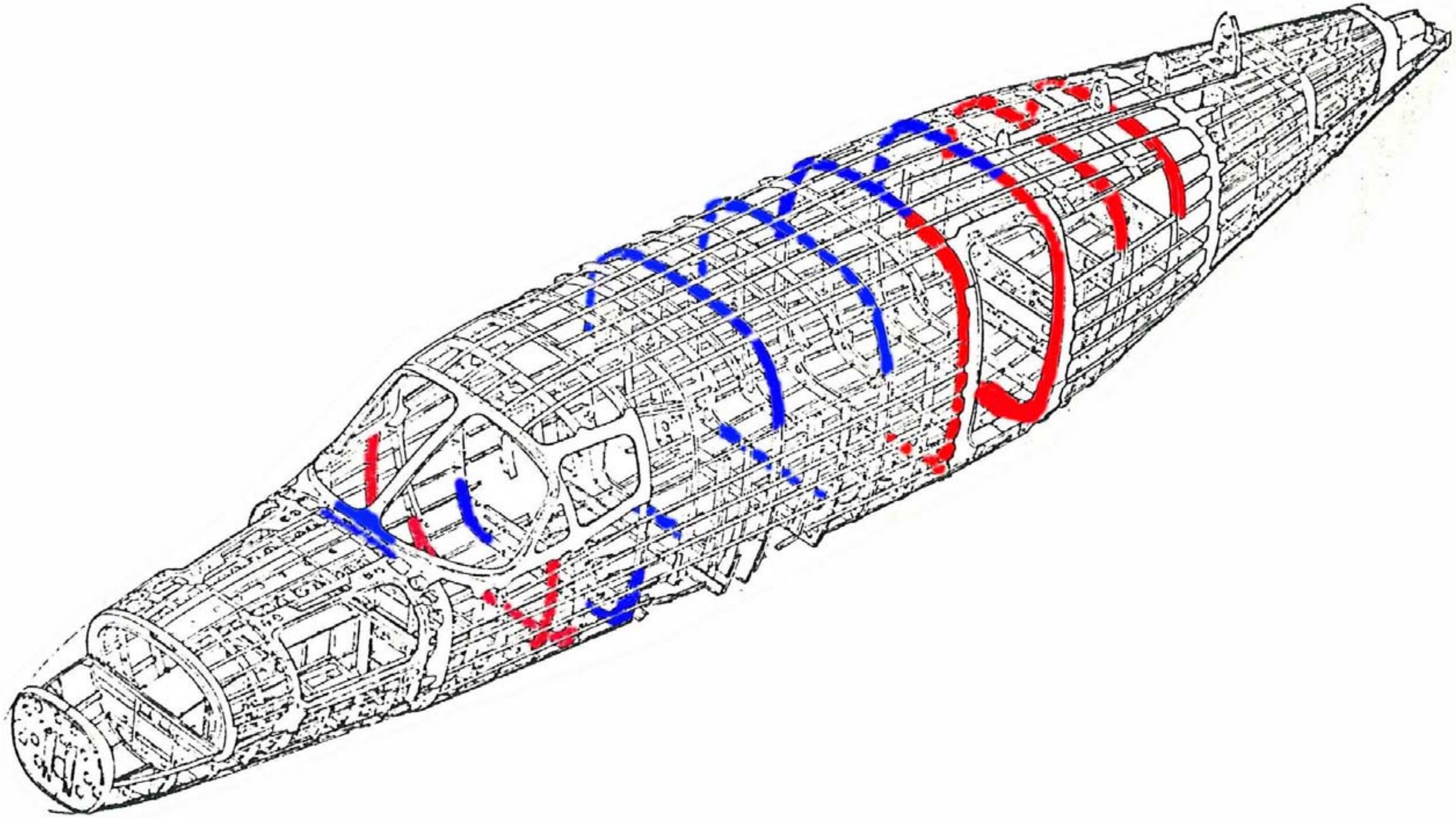
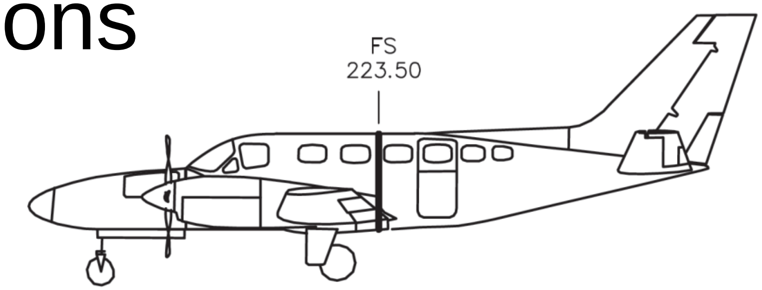
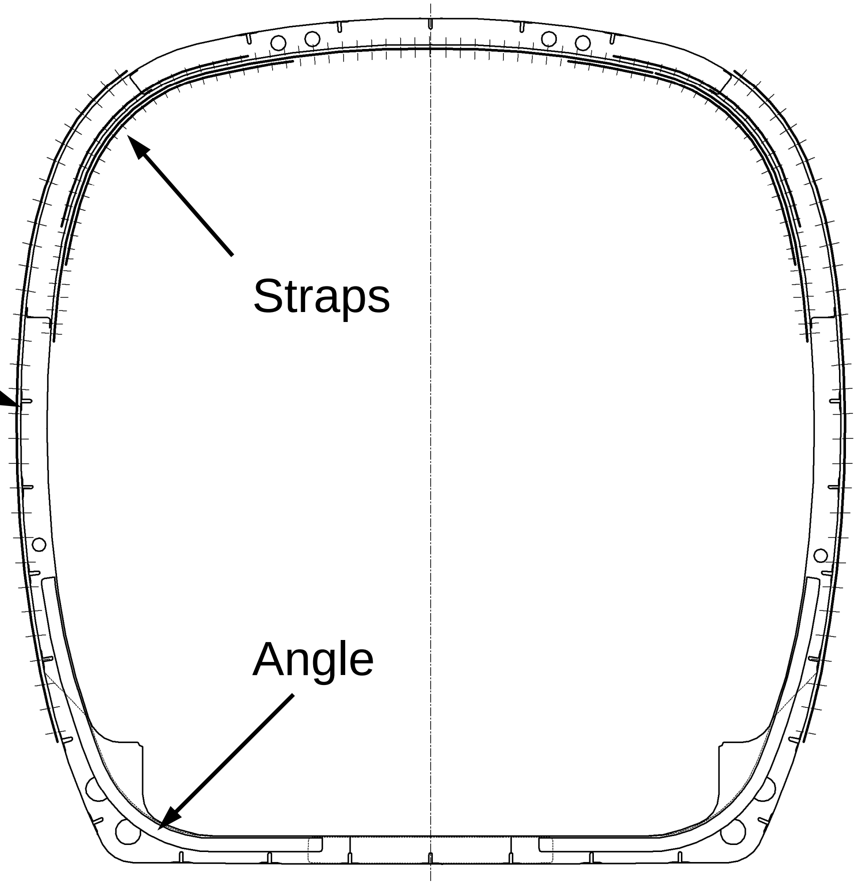
Critical fuselage frames



Fuselage frame modifications



- A typical frame is at FS 223.5.
- Reinforcing is added at the top and bottom corner inner caps, and on the outer cap on the side



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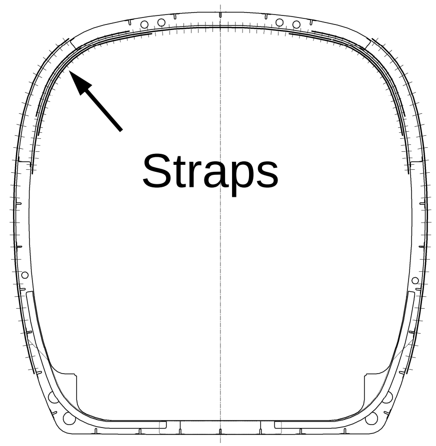
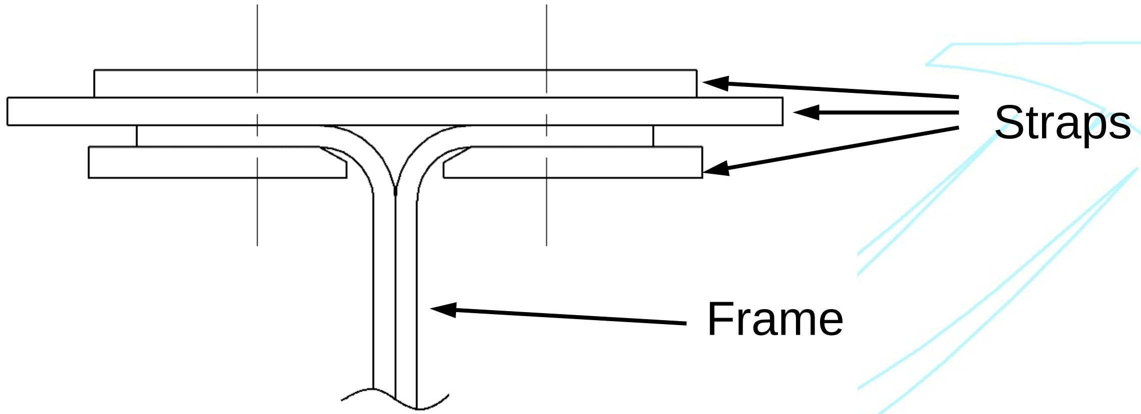
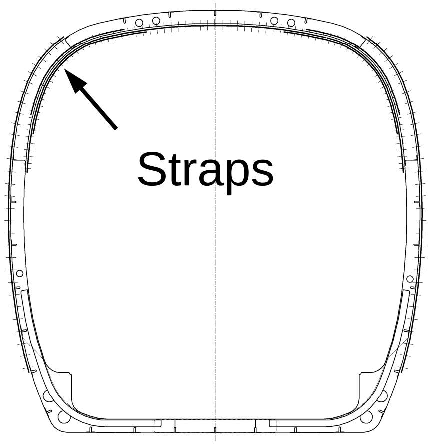
Frame modifications - example

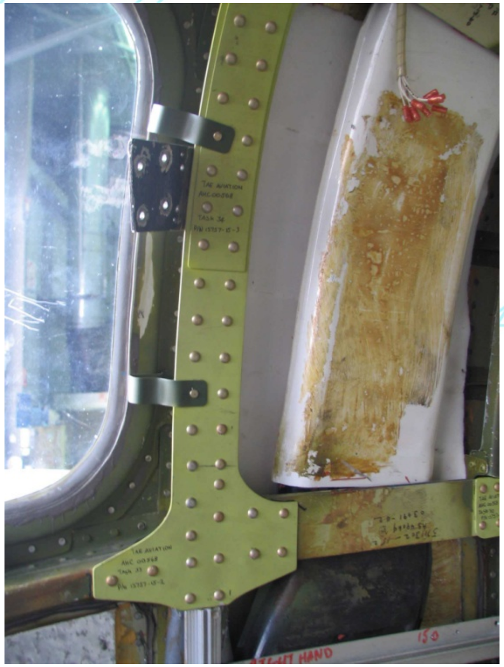
- Section of straps at top corner
- Original section had a single strap
- Second strap is wider, so edge can be inspected for cracking





Frame modifications - example





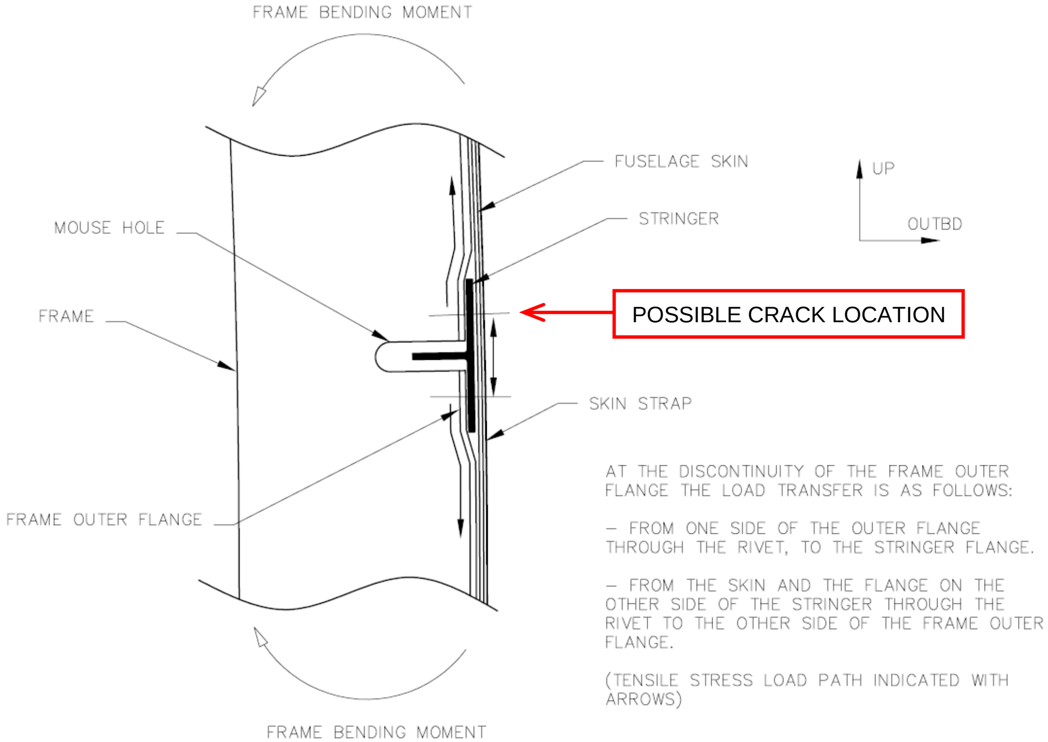
Fuselage side failsafe strap

The outer flange and skin on the fuselage side is in tension.

Frames are notched to clear stringers. The skin acts as the outer flange at these locations

A possible failure scenario is a skin crack along the rivet line to a stringer. This crack would result in failure of the frame.

A strap has been added to the fuselage sides at the highly loaded frames to make the frame failsafe.



Damage tolerance analysis

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Damage tolerance analysis has been done using AFGROW.

Crack growth life was calculated at each critical location.
Residual strength was taken as limit load.

Assumptions for initial crack size are generally:

Primary crack

For existing structure: a = 0.080

This is the size which is detectable at the time of modification.

New structure: a = 0.050

Secondary crack

All structure a = 0.005

Damage tolerance analysis

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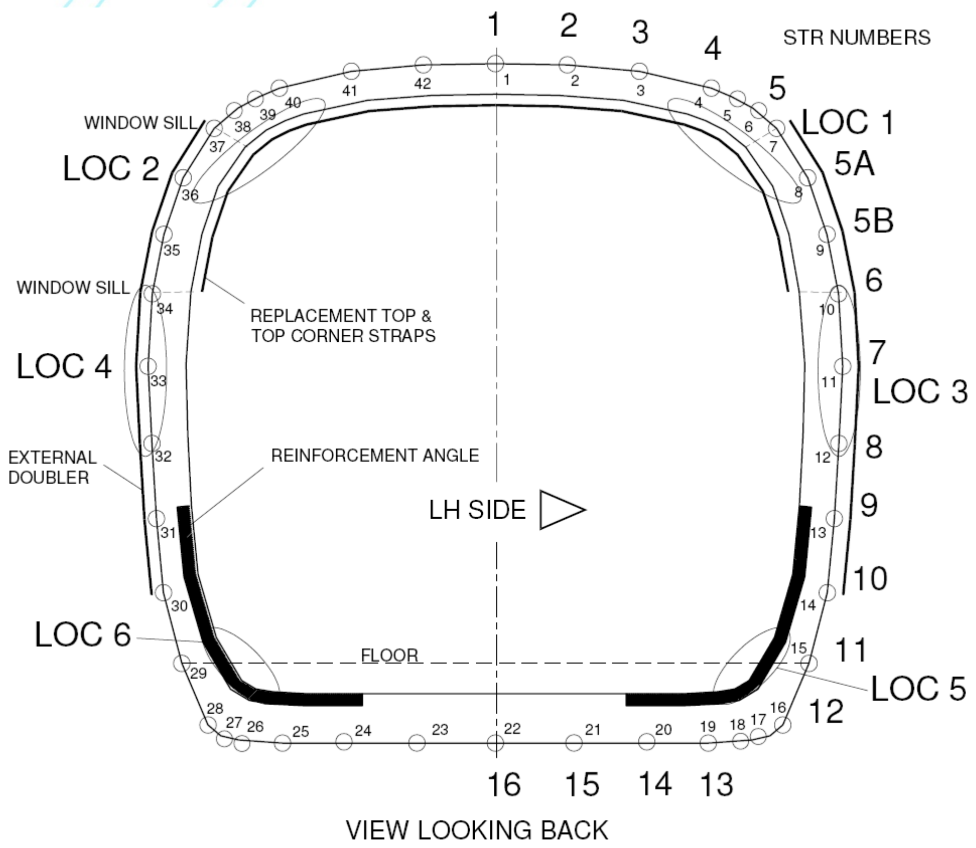
Frame 223.5 analysis locations are shown



Damage tolerance analysis

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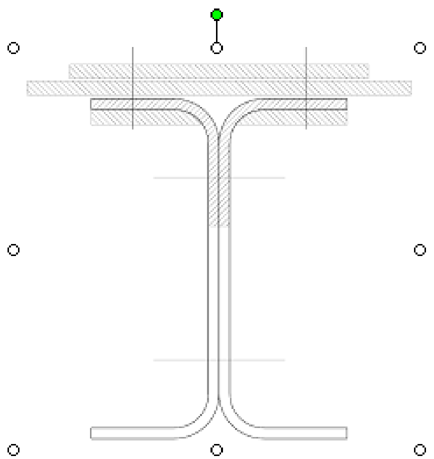
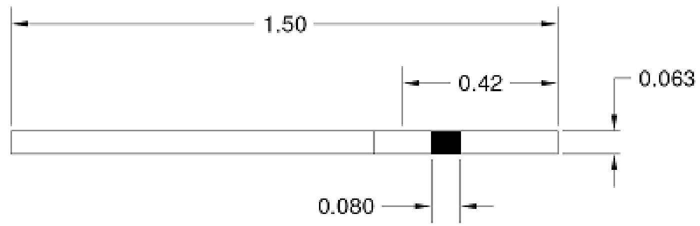
Location 1 & 2 - Top corner inside flange strap

LOC 1 & 2

Modified
Configuration



Crack Growth Model



Damage tolerance analysis

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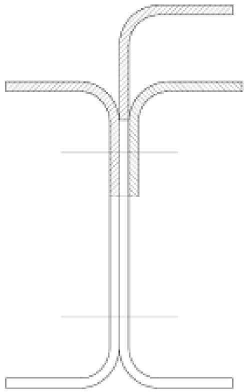
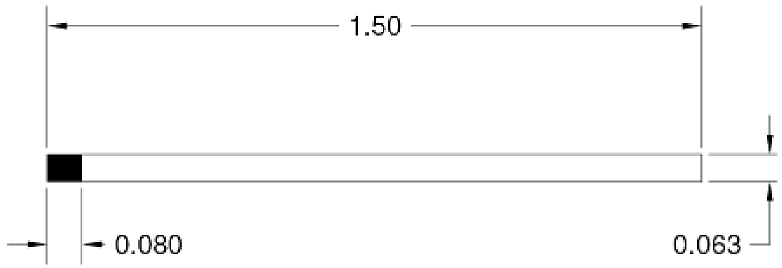
Location 5 & 6 - Bottom corner inner flange angle

LOC 5 & 6

Modified
Configuration





Crack Growth Model

Damage tolerance analysis

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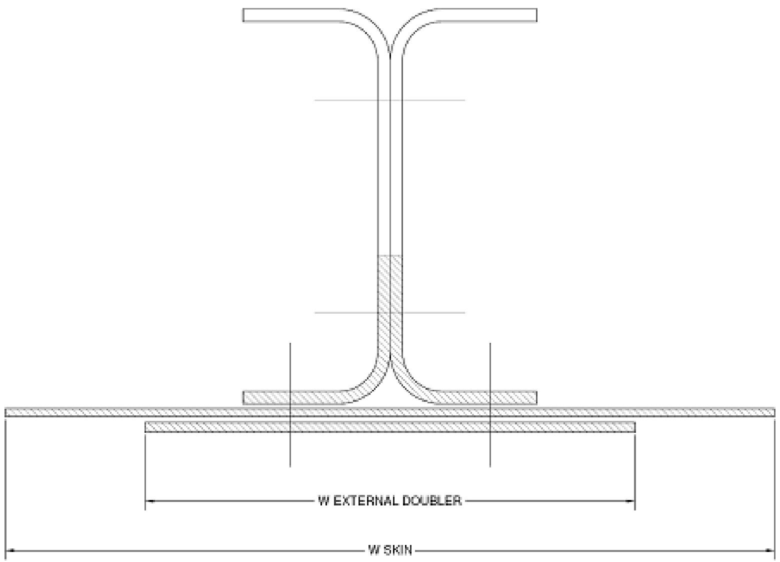
Location 3 & 4 - Side external strap

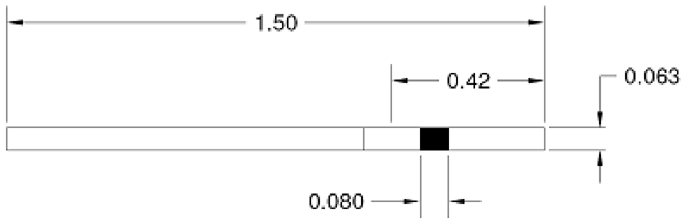
As well as the AFGROW crack growth analysis, calculations were done for long cracks in the skin at the strap. These analyses used a detailed model.

LOC 3 & 4

Modified

Configuration





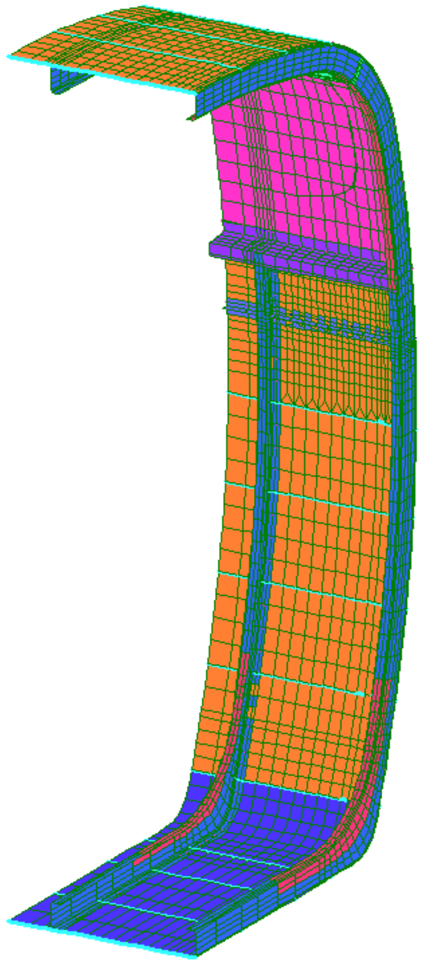
Crack Growth Model

FEM - Detail Model of Frame

A detailed, **fine mesh model** of a fuselage frame and skin was made.

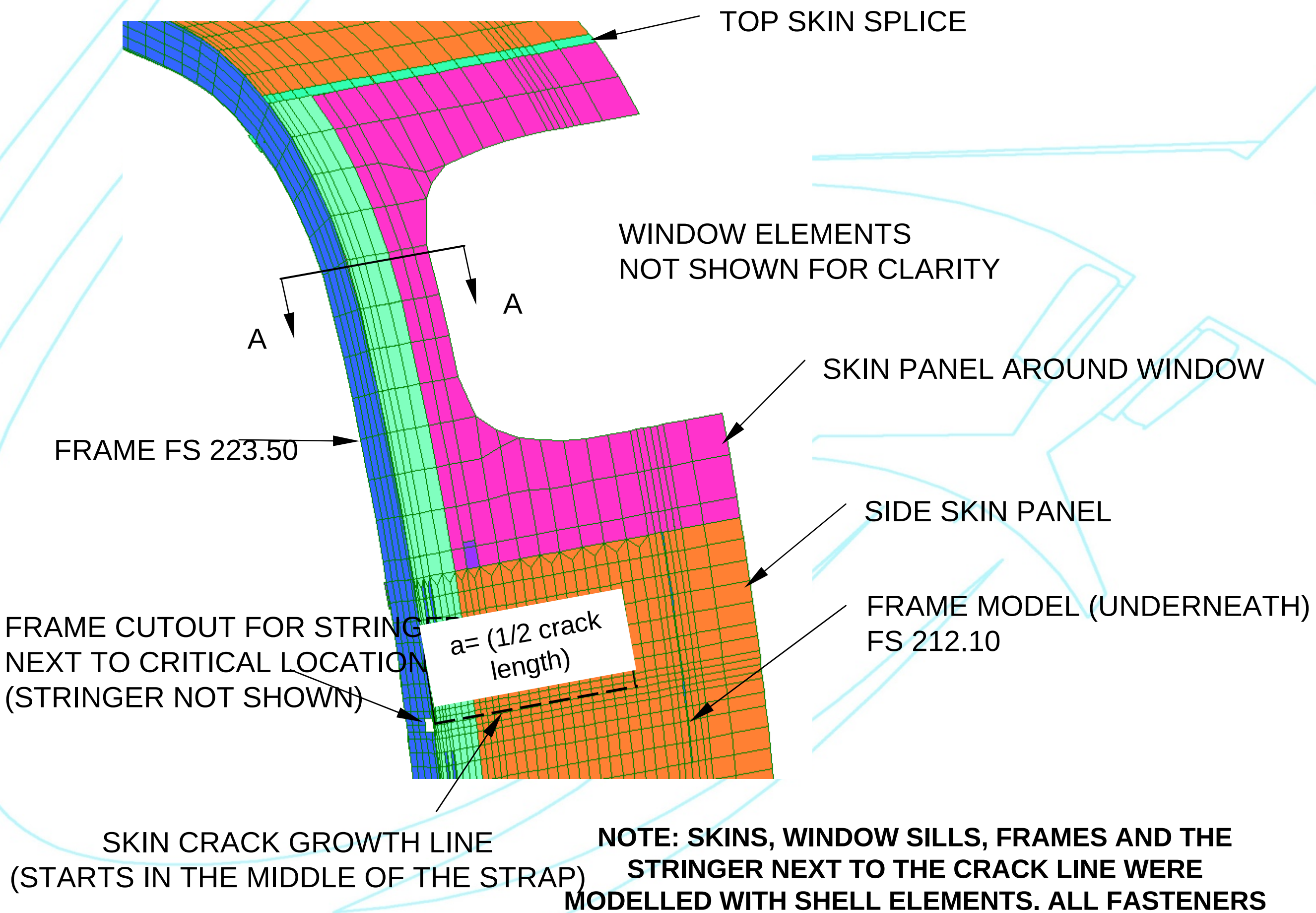
This model had two functions:

- Straps were added on the side of the fuselage to make the frames damage tolerant. The model is used in the prediction of crack growth and residual strength near the straps.
- The coarse mesh main model does not give accurate stress results in the outer flange of the frames, where the skin radius is changing rapidly. At these locations load is shearing from hoop tension in the skin to tension in the frame cap. The fine model captures this effect and gave a check on stress from the main FEM.



FEM - Detail Model of Frame

Fine mesh model of fuselage frame and skin



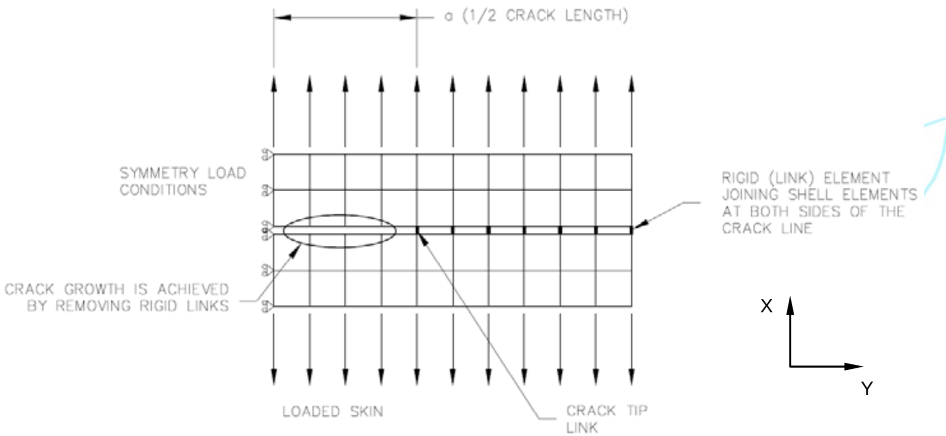
NOTE: SKINS, WINDOW SILLS, FRAMES AND THE STRINGER NEXT TO THE CRACK LINE WERE MODELLED WITH SHELL ELEMENTS. ALL FASTENERS BETWEEN MEMBERS WERE MODELLED WITH SPRING ELEMENTS.

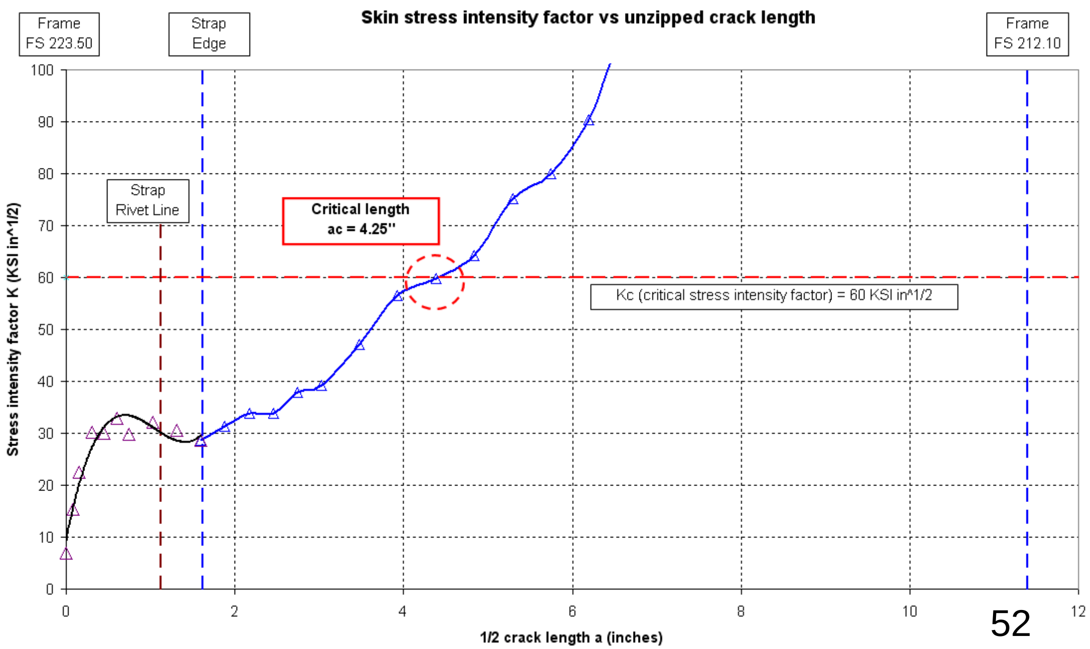
FEM - Detail Model of Frame

The stress intensity for a long skin crack was calculated using the **crack closure technique**.

This enabled calculation of crack growth rate and residual strength.

The critical crack length was long enough to be detected from external inspection





Forward pressure bulkhead

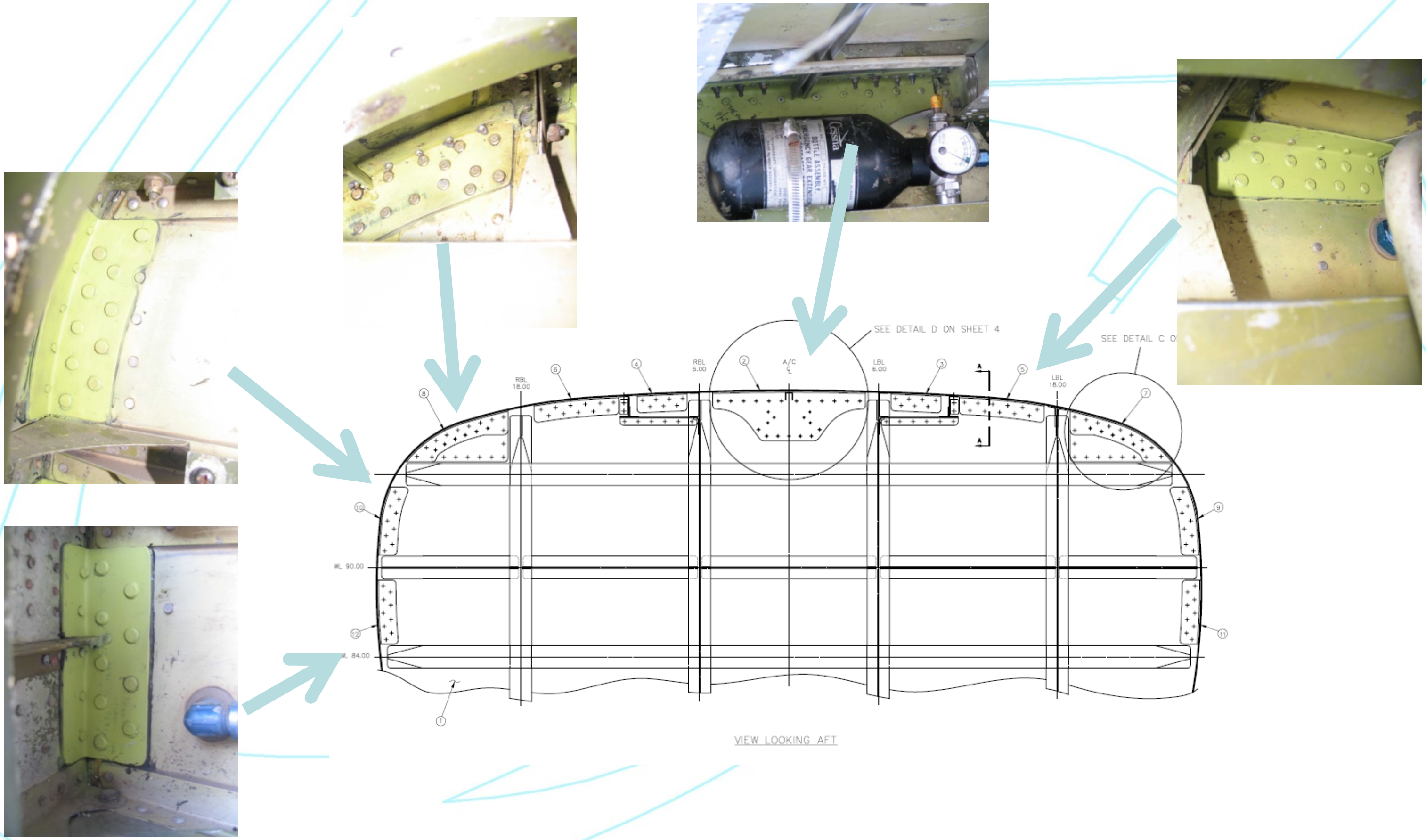
The forward pressure bulkhead is flat, and is located level with the forward edge of the windscreen.

A common area for cracking is the angle between the bulkhead and the skin at the top of the bulkhead.

It is very difficult to inspect from behind, since it is covered by the instrument panel.

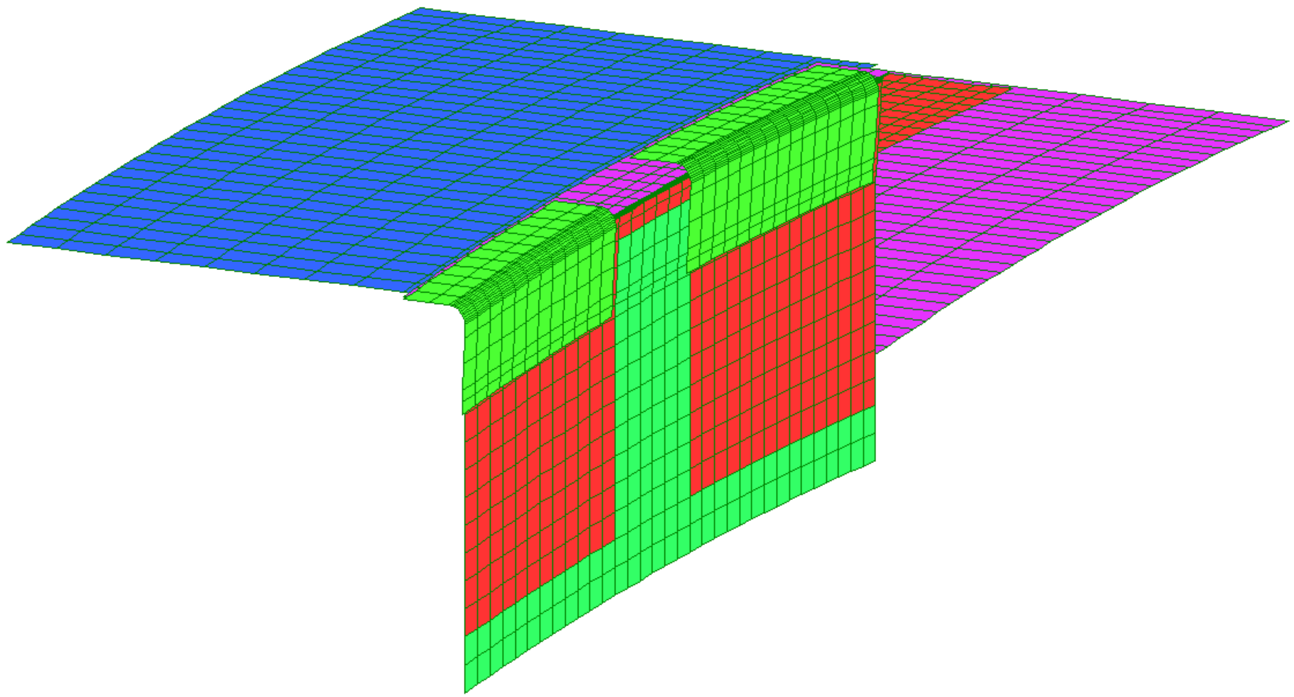
Forward pressure bulkhead

The angle at the top of the frame is replaced by a modified angle.



Forward pressure bulkhead

A **fine model** of a section of angle was made.
This enabled justification of a modified angle.



Fuselage bottom stringers

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The bottom of the fuselage is flat.

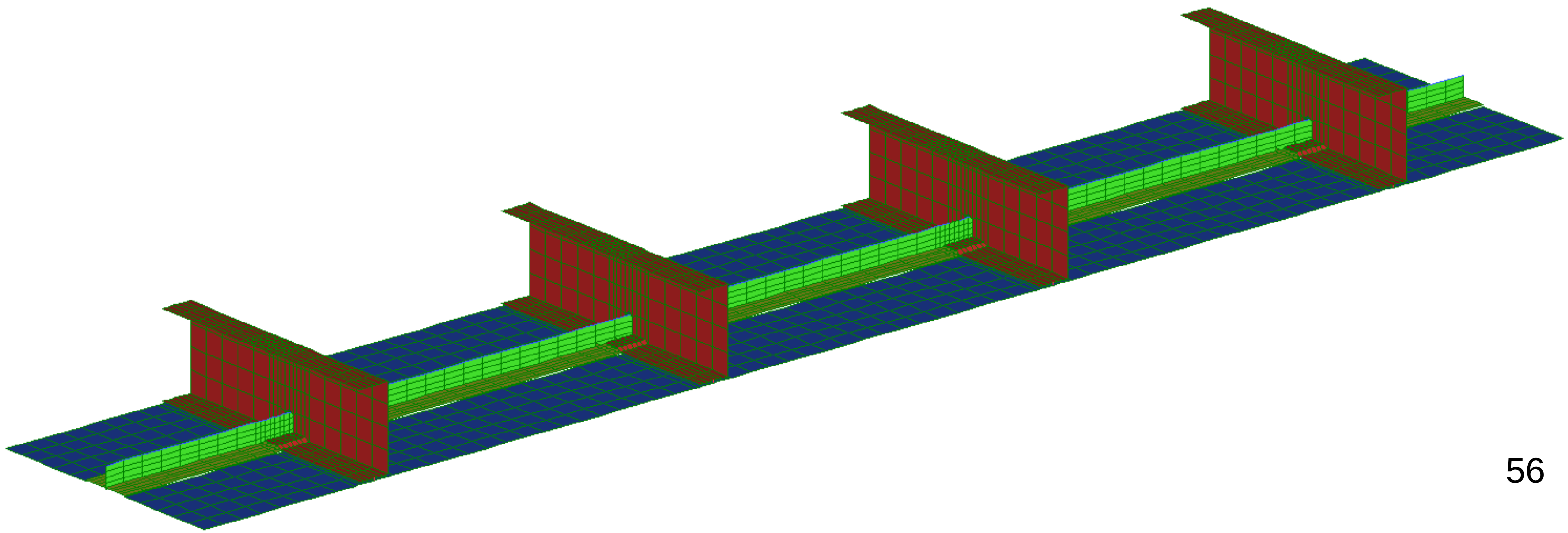
Pressure loads cannot be taken by hoop stress in the skin.

Pressure on a panel is carried by membrane and bending stress to the adjacent frames and stringers.

Stringers act as continuous beams, supported at the frames

A **fine model** of a stringer was made to calculate stringer bending stress.

Nonlinear solution gave correct membrane stresses in the skin, and hence load transfer to the stringer

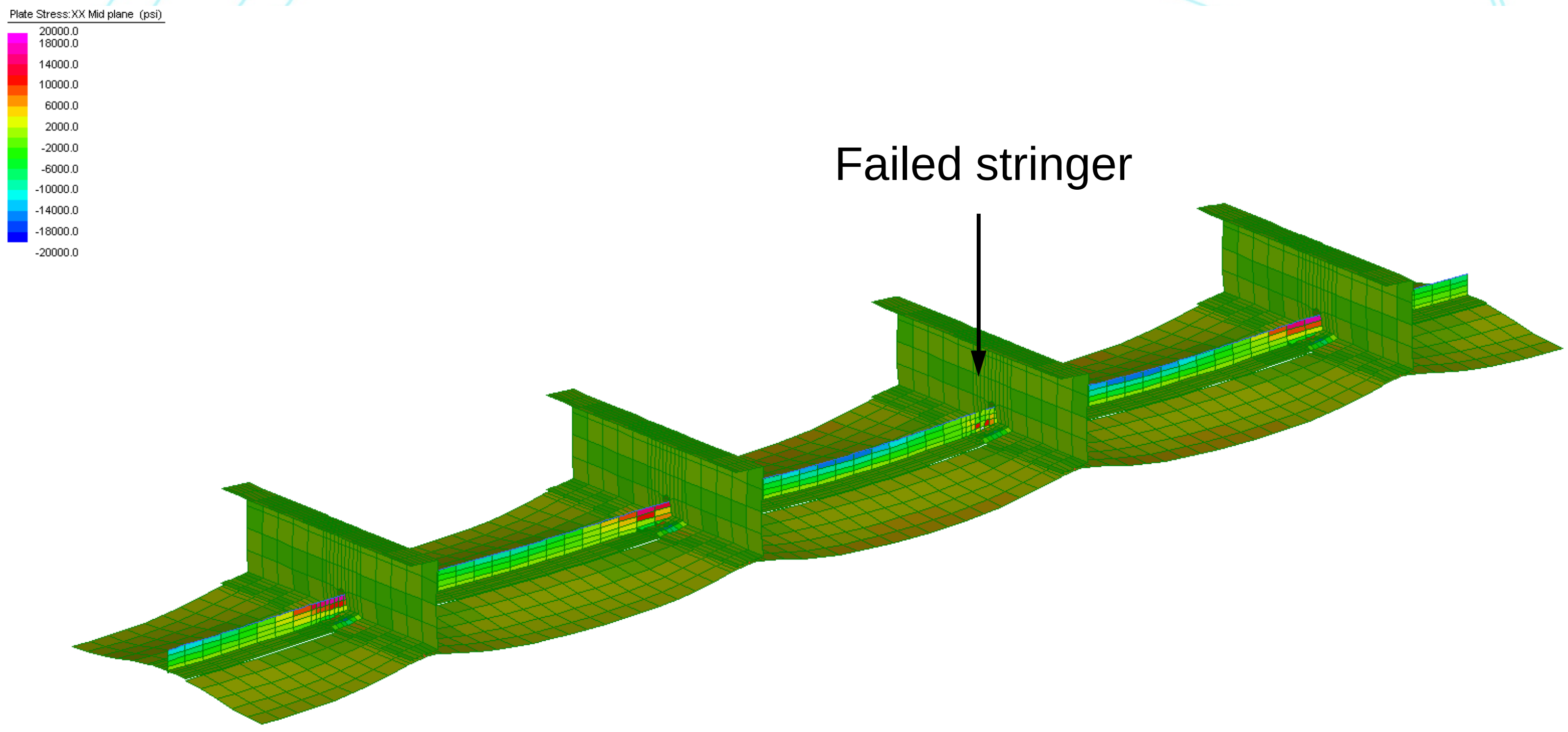


Fuselage bottom stringers

Stringer stress is maximum at the top of the stringer, where it passes through the frames.

There are no stress concentrations at the top of the stringer, but the stress is high enough to give a relatively short life.

The structure is failsafe, since the stringer is now simply supported at the frame. This failure scenario is shown below



Fuselage strain gauge testing

The FEM is validated by comparing results to strain gauge tests.

Fuselage
- Ground test
- Pressurisation

Some revised inspections

- Forward pressure bulkhead
- Fuselage frame inspections
- Wing to fuselage attachment
- Tailplane inspections
- Flap debonds
- Wiring

Conclusions

Wing safe life to 40,000 hours
Empennage safe life to 40,000 hours
Some additional inspections for wings, empennage, control surfaces
Fuselage modifications required for 40,000 cycles life
New IFCA required
STC in final stages of approval

Questions