Piper

Recent PA-46-600TP NL Modelling Applications

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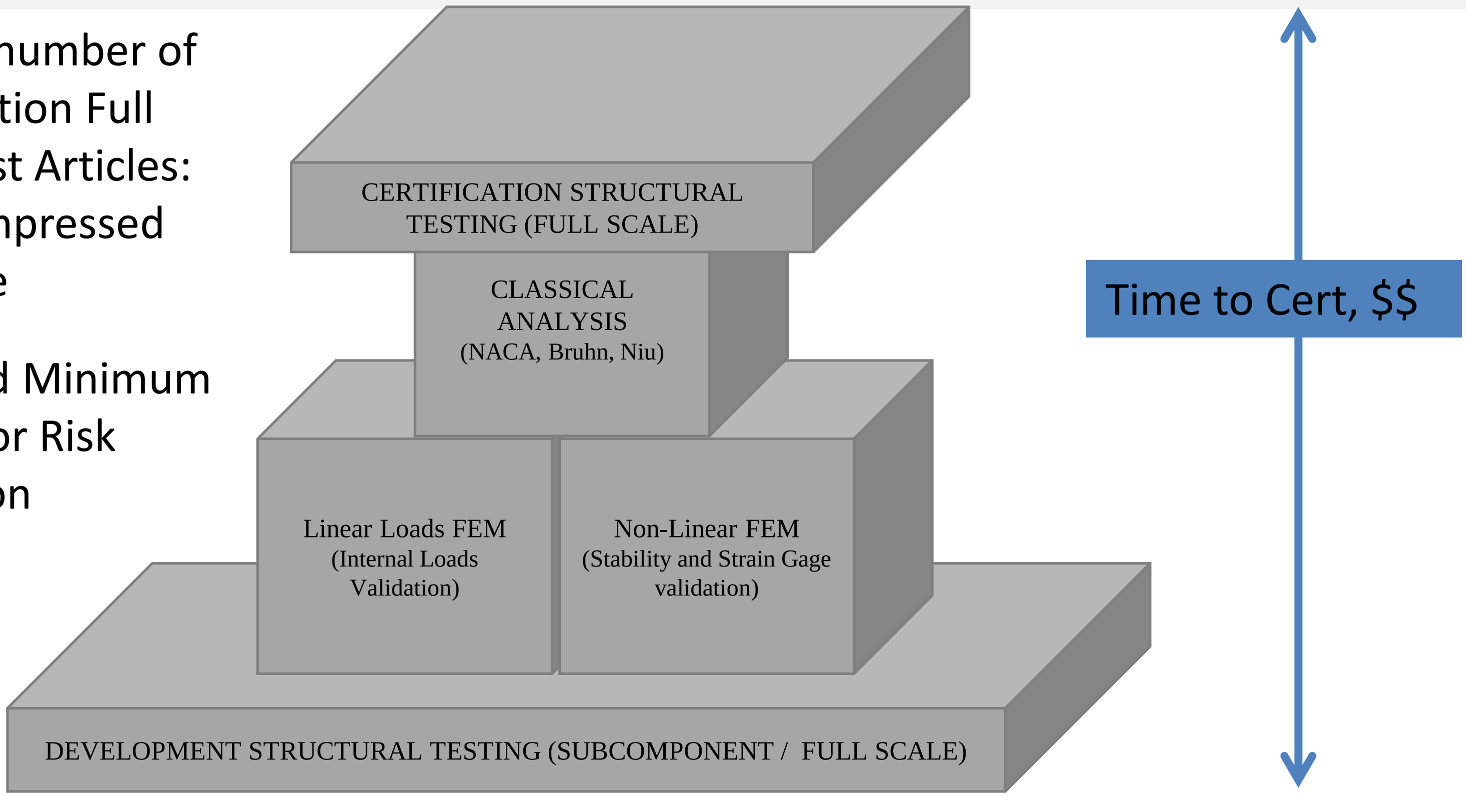
FAA Accepted Means of Compliance
Analyses Supported By Test : Lightweight Aircraft Structures

- Limited number of Certification Full Scale Test Articles: fully compressed schedule
- Required Minimum Efforts for Risk Reduction



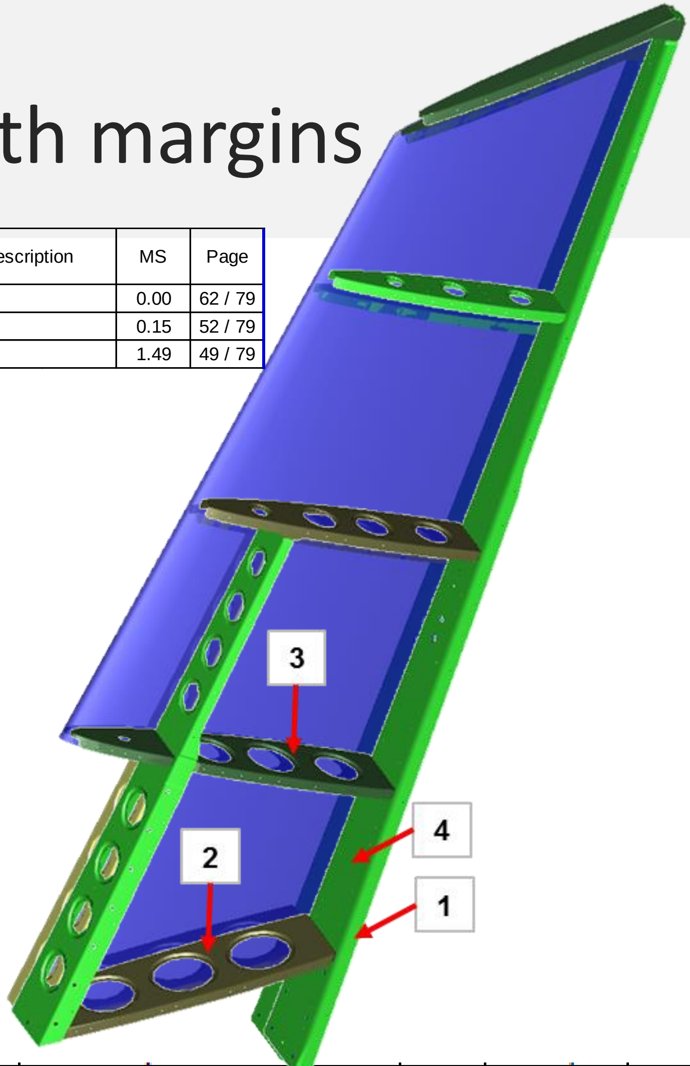
- *Removal of ANY block effort risks re-design, re- certification efforts and ultimately risks program cancellation.*

PA-46-600TP
Case Study: Vertical Tail Stability Driven Analysis

- Linear Analysis
- NL-FEM of Test Configuration
- Development Test
- Design Changes Incorporated
- NL-FEM of Revised Configuration
- Spar, Rib, and Attachment Margins
- Joint Analysis Using NL-FEM Results

– linear analysis strength margins

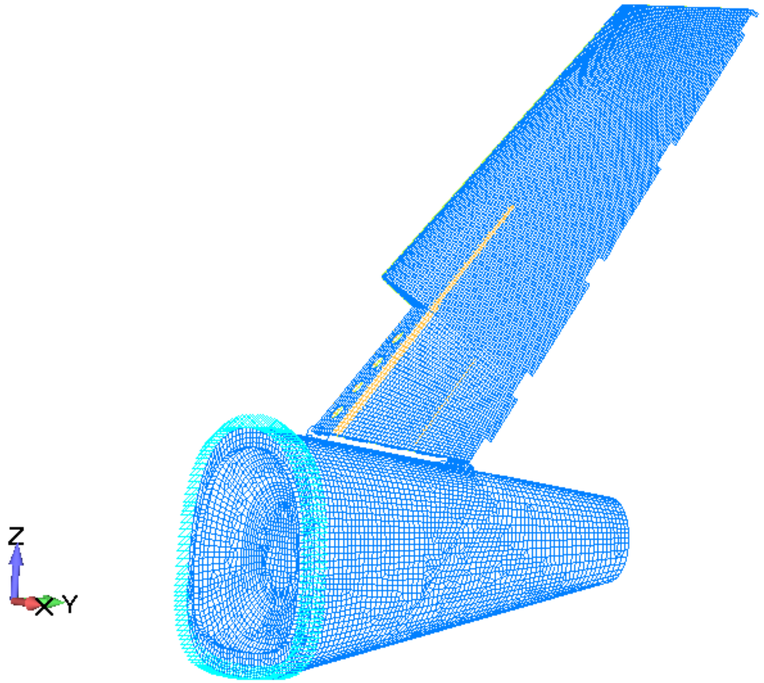
Loc. ID	Component	Critical Check	Load Case	Load Description	MS	Page
1	Aft Spar	Crippling	29		0.00	62 / 79
3	Rib 2	Buckling	29		0.15	52 / 79
4	Aft Spar Web	Buckling	29		1.49	49 / 79



ST1	Strength	368(L)-1	2/5	Lateral Gust	V. Stab. (Bending)	Limit	N/A	0.0	0.00	443

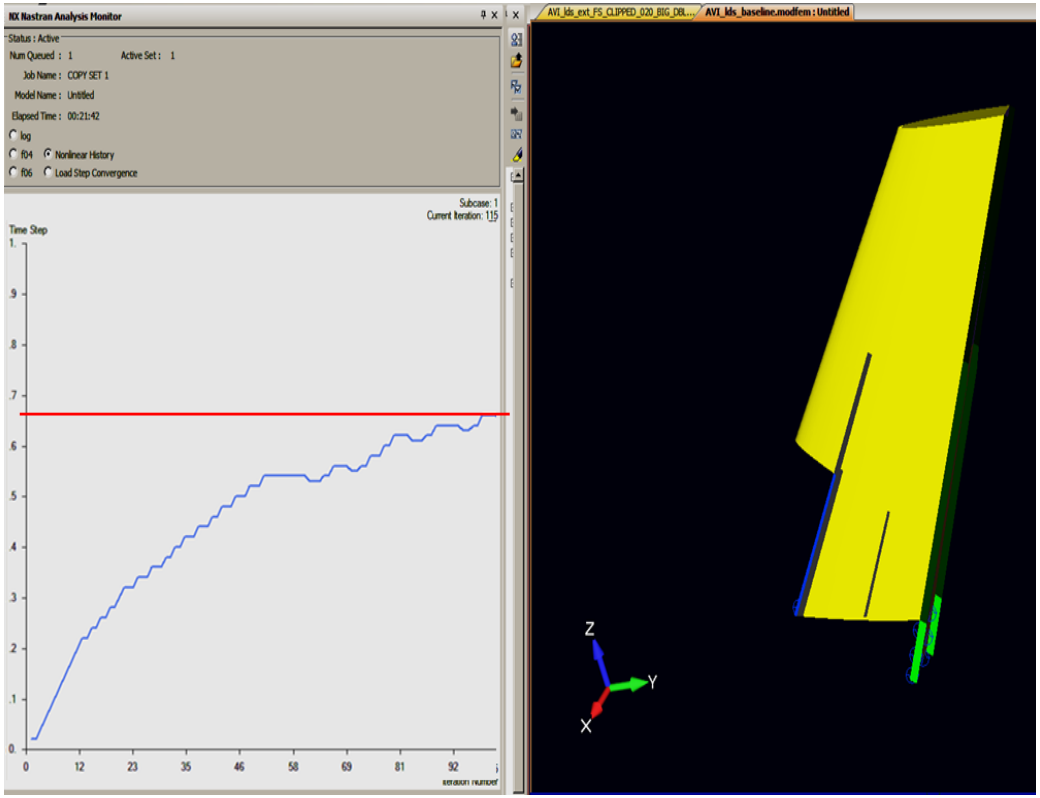
Vertical Tail development NL model

- 38,622 nodes, 41158 elements
- Includes representative tail-cone to pressure dome



Convergence plot – test configuration

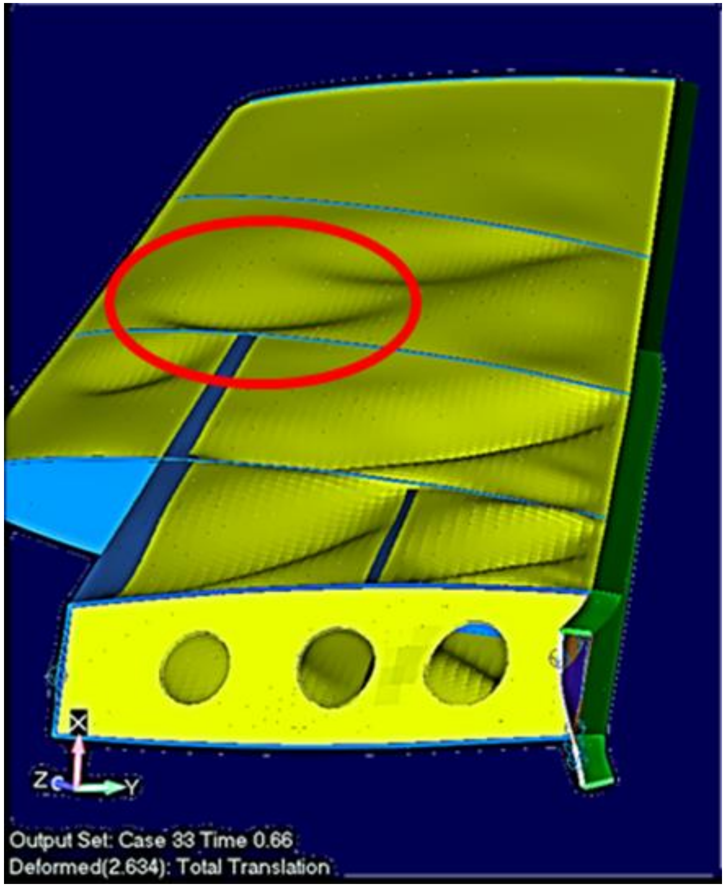
Final time step = 0.66 (118.5%)



Non-linear fem results

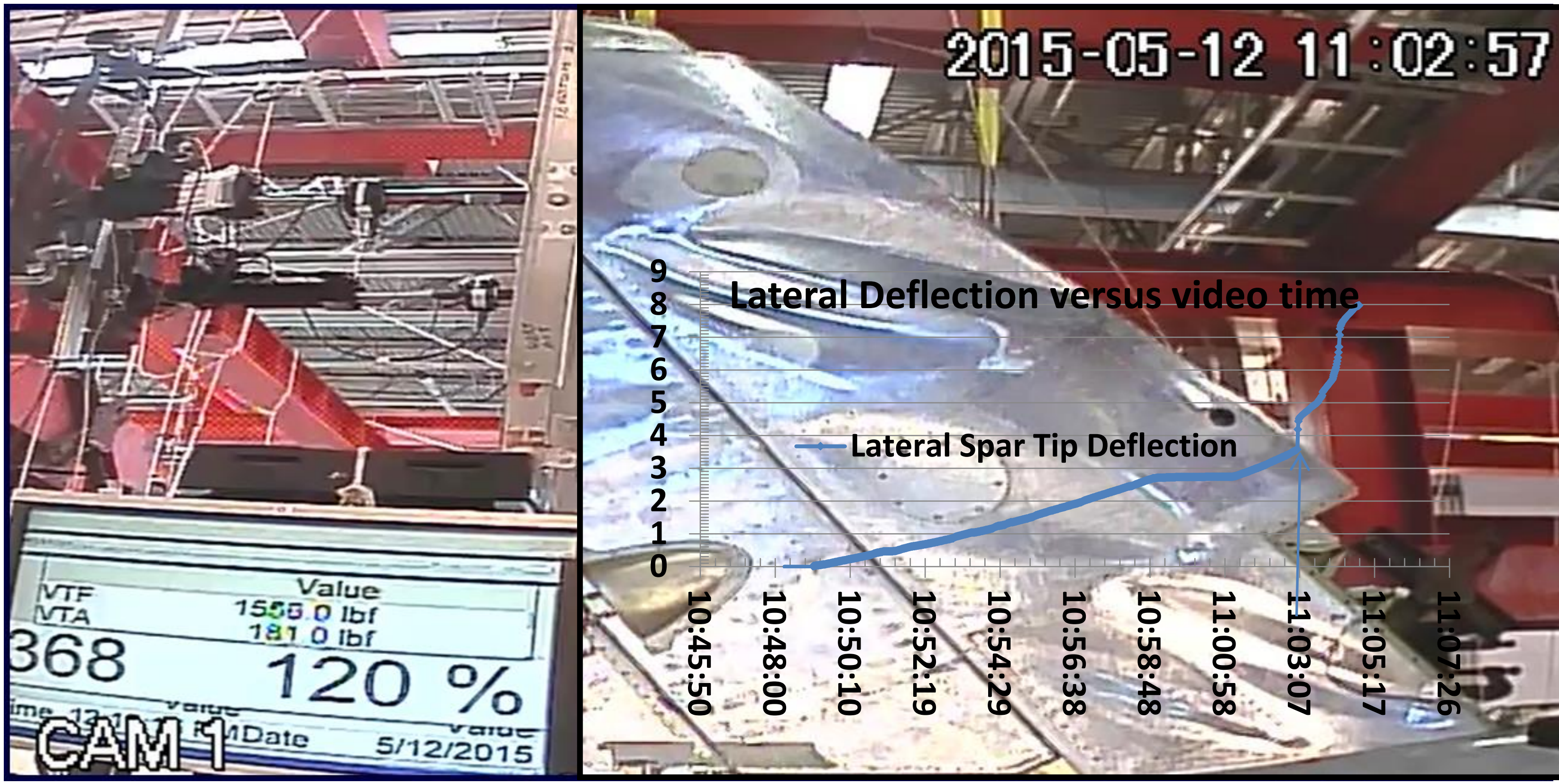
- development configuration FEM (AVI_lds_baseline) fails to continue finding a solution at time step 0.66

  0.66 × 180% = **118.8%**



Development Test

Development Test-



Cause and Analysis Observations (Ref. VB-25XX)

9.2.1.1.1 Failure Cause

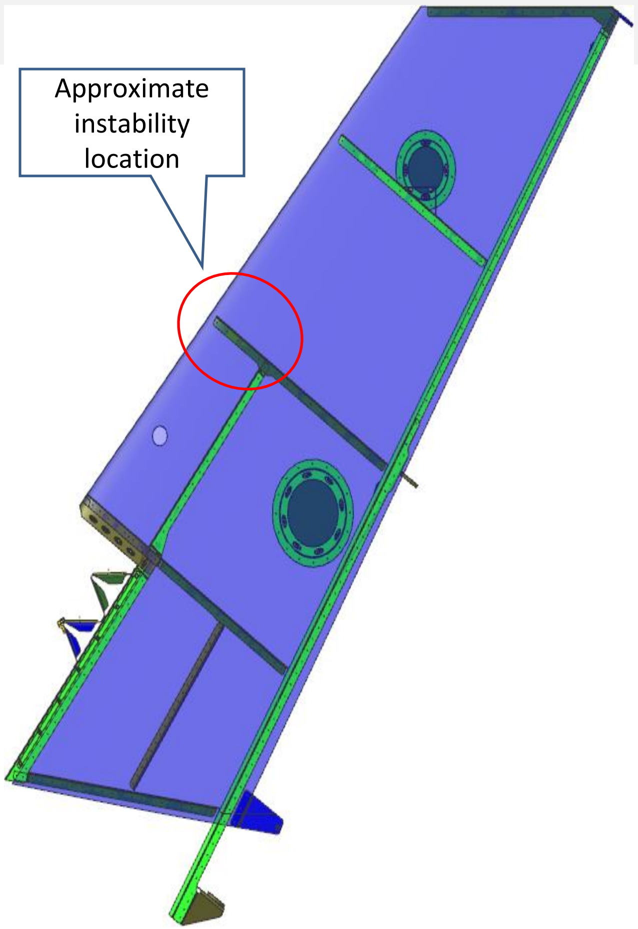
- Vertical stabilizer was capable of carrying approximately 118% of limit load for NREG 368 for 3 seconds.
- Root failure was instability of the forward skin leading edge radius, just outboard of the forward stub spar.
  - Original analysis was primarily strength based and did not evaluate the stability of the leading edge skin nor secondary effects of buckling.
- Forward spar, rear spar and ribs were intact and not failed at the point of instability.

Observations:

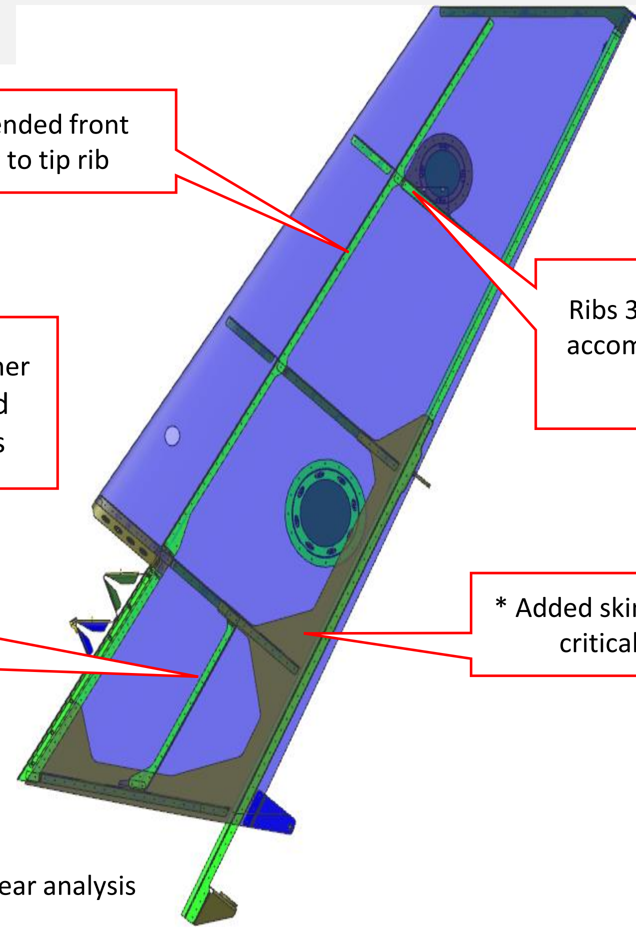
- Non-linear large displacement analysis can be predictive of overall instability for shell type structure
- Deformed shape review is **critical** for determining whether non-convergence is driven by an overall instability or something structurally bounded to allow for higher loading levels.
- Stress and Strain levels are just as important at any time step but may take a back-seat to stability driven failure and these levels can substantially change over very few time steps.

Key design changes

83401-012 (Tested Design)



83401-013 (Revised Design)

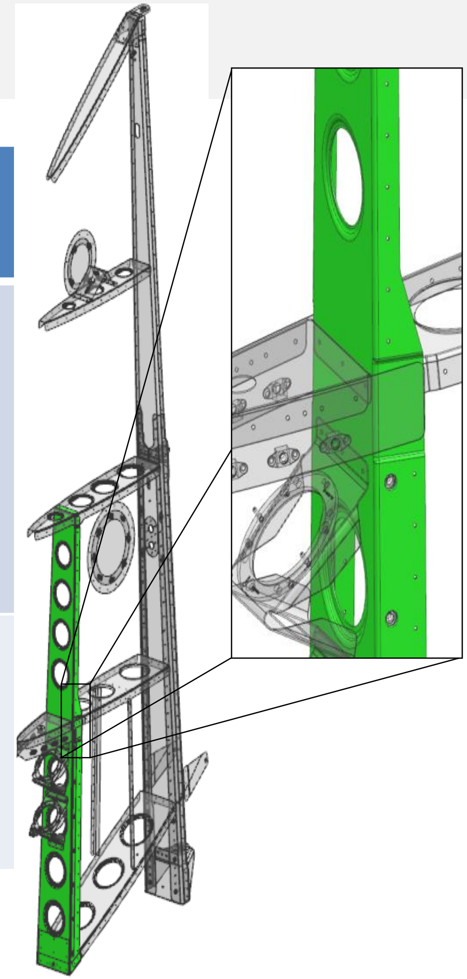


* Changes required per non-linear analysis

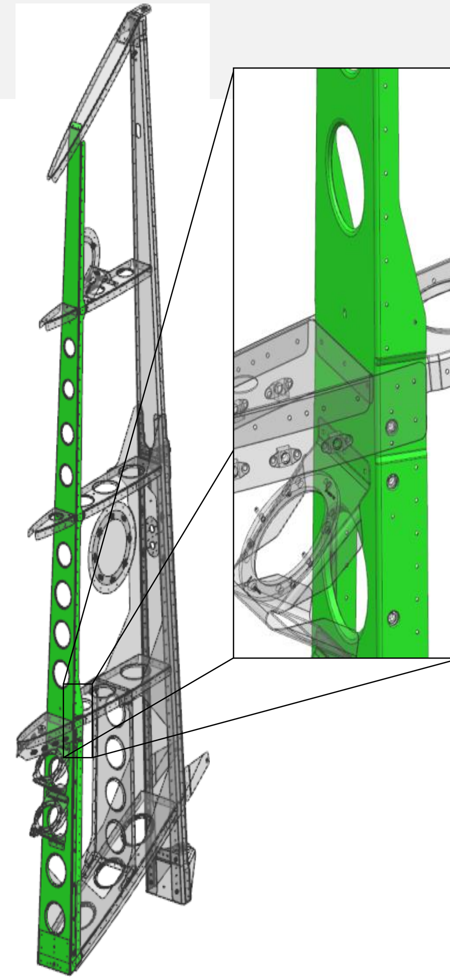
Revised design – front spar

Revision	Reason
Extended spar to tip rib	Full spar needed to complete torque box for high torsion load cases
Increased flange width near Rib 2	To further distribute fastener loads at the joint

83414-002



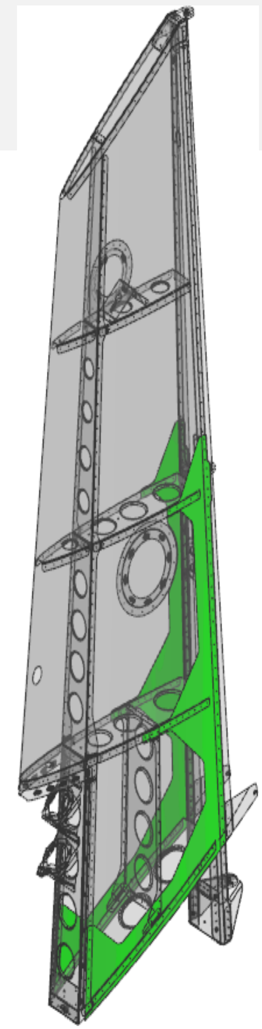
46V550012-001



Revised design – skin Doubler

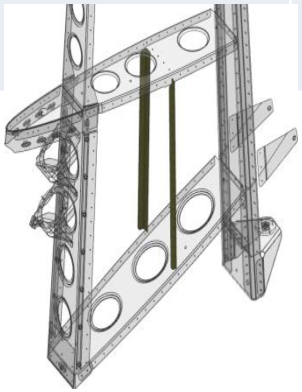
Revision	Reason
Created .020" bonded doubler (both sides)	Required for peaking stresses due to buckling at rib-to-spar interface
	Effective skin thickness increase required for fastener loads
	Locally stiffens skin panels that are subject to shear buckling

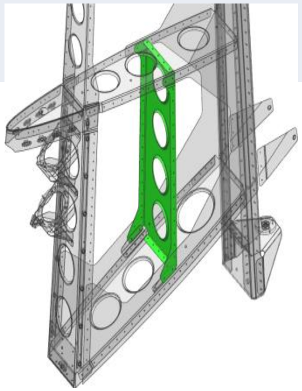
46V550027



Revised design – intercostal

Revision	Reason
Created intercostal to replace skin panel stiffeners	Previous testing revealed that panel buckles spread across existing stiffeners around limit load-nodal line not enforced (see photo)
	Intercostal offers greater resistance to shear panel buckling

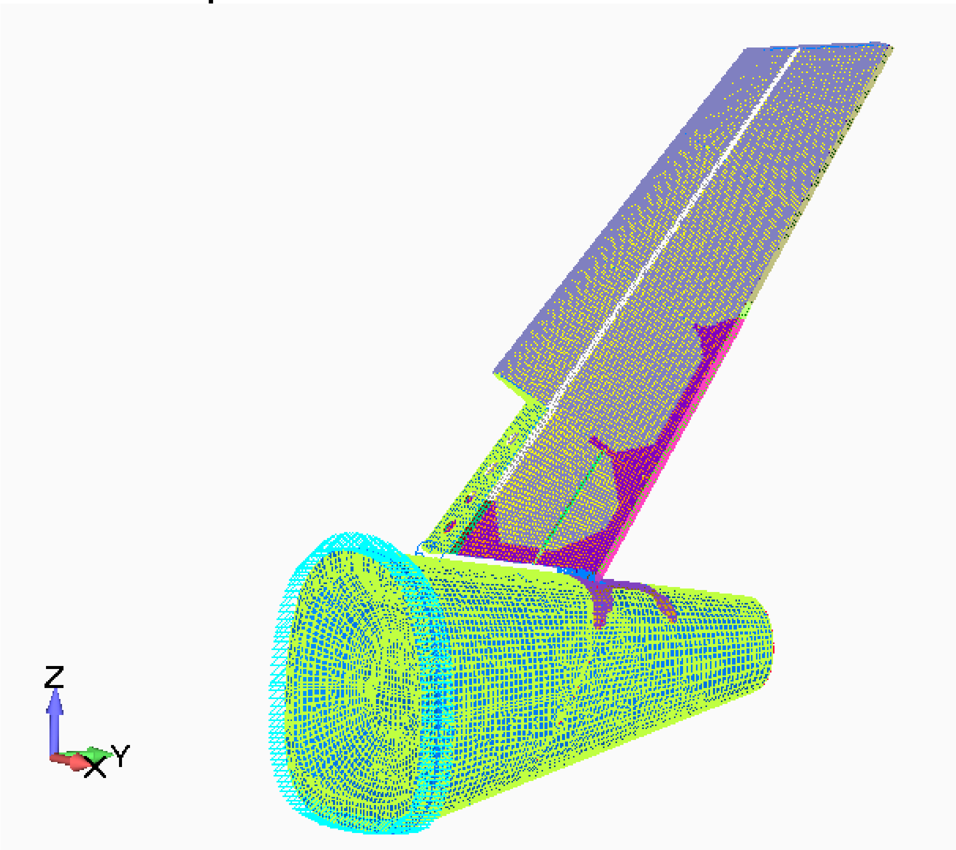




46V550025

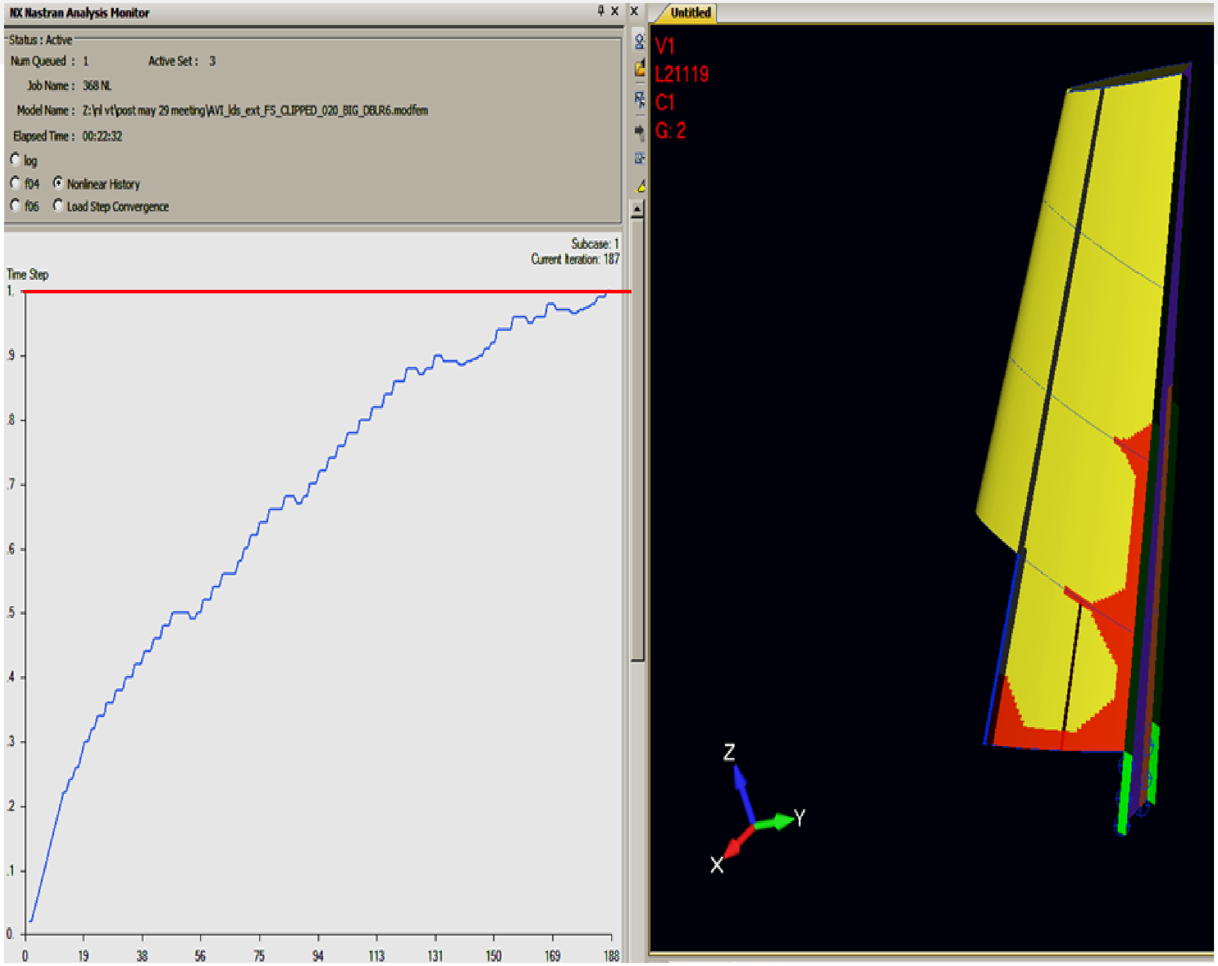
Vertical Tail production design NL model

- 41959 nodes, 44310 elements
- Includes representative tail-cone to pressure dome



Convergence plot – revised design for production

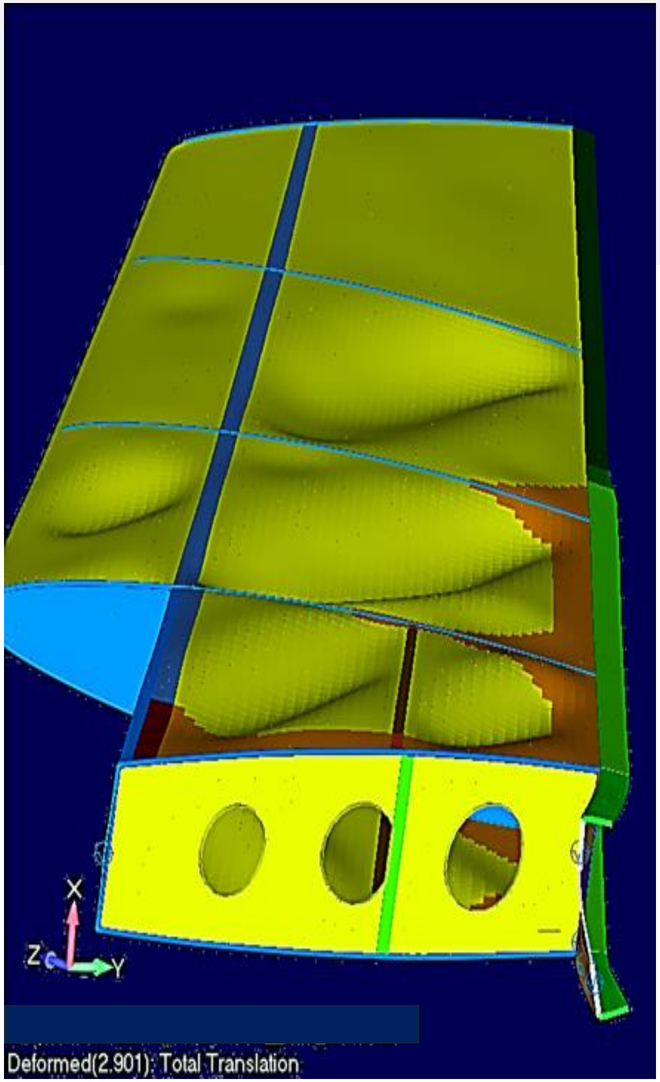
Final time step = 1.00
(>150 DLL%)



Non-Linear Rem Results

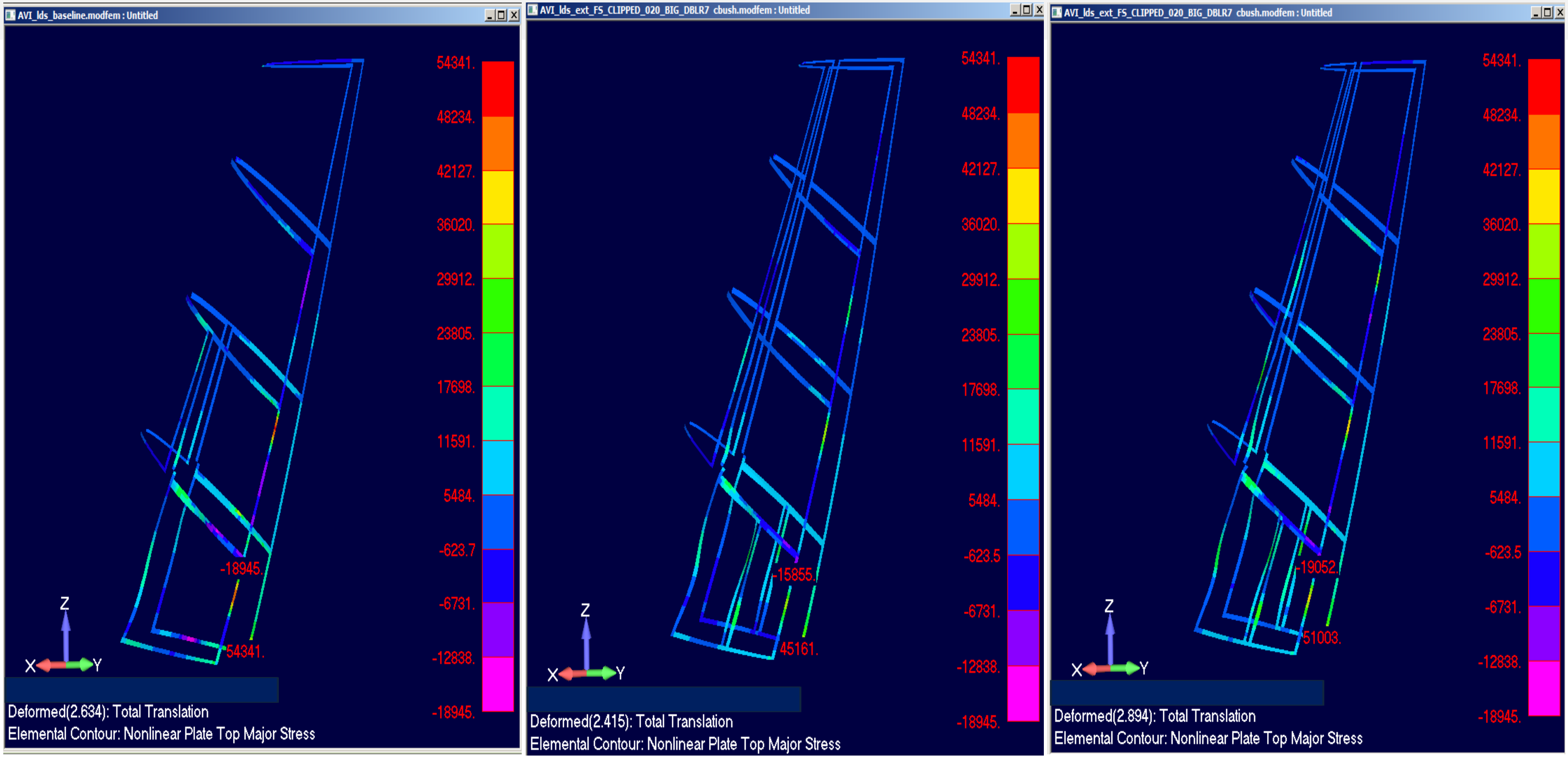
- Revised configuration FEM converges through time step (151.2% DLL and beyond)

- Revised VT is expected to be stable at 150% Limit Load



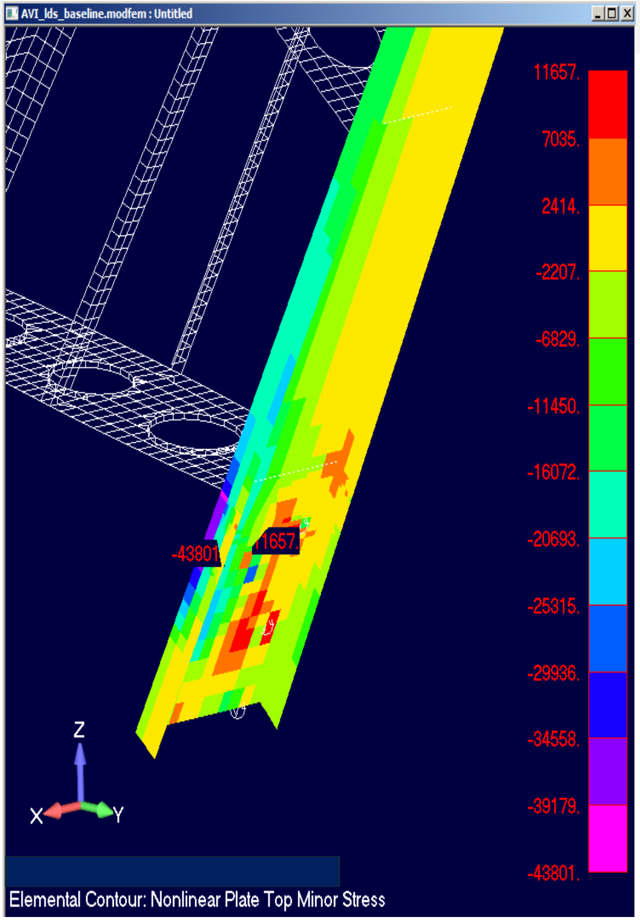
Non-Linear FEM Results

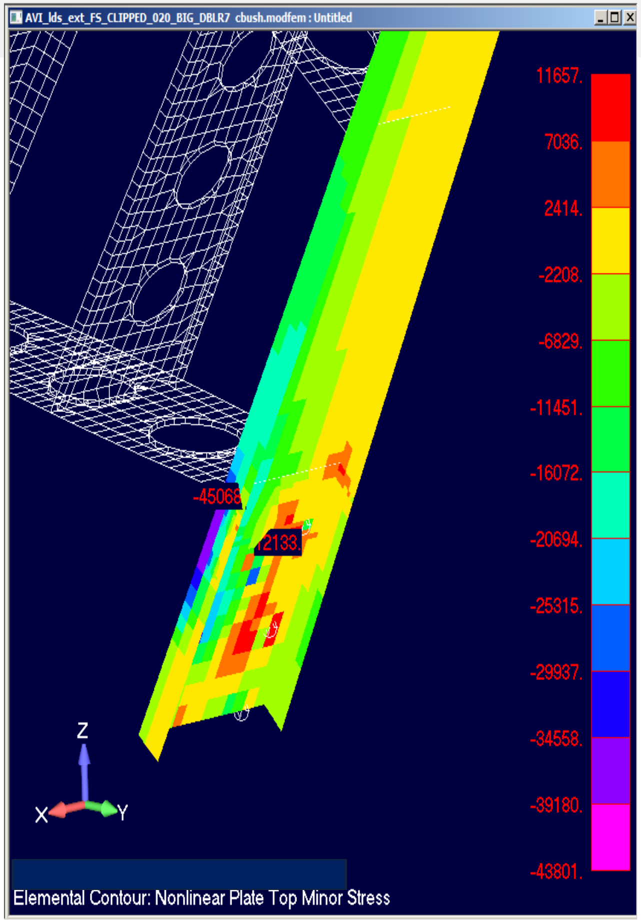
model for production design with skin edge doubler shows reduced Plate Top Major Prin Stresses at 151% compared to Test model at 119%



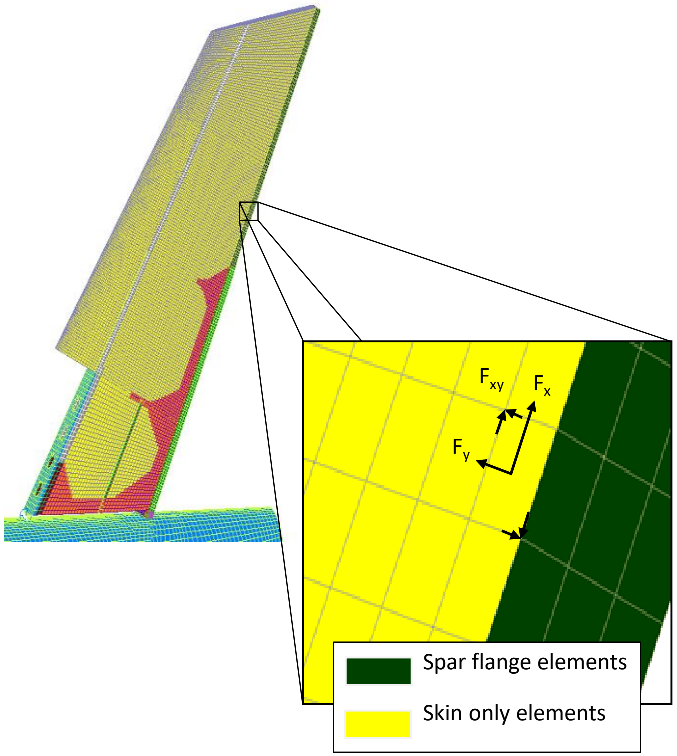
Non-Linear FEM Results

Development Test model and Revised production model show similar stresses near VT-fuselage attachments at 119% (as expected)

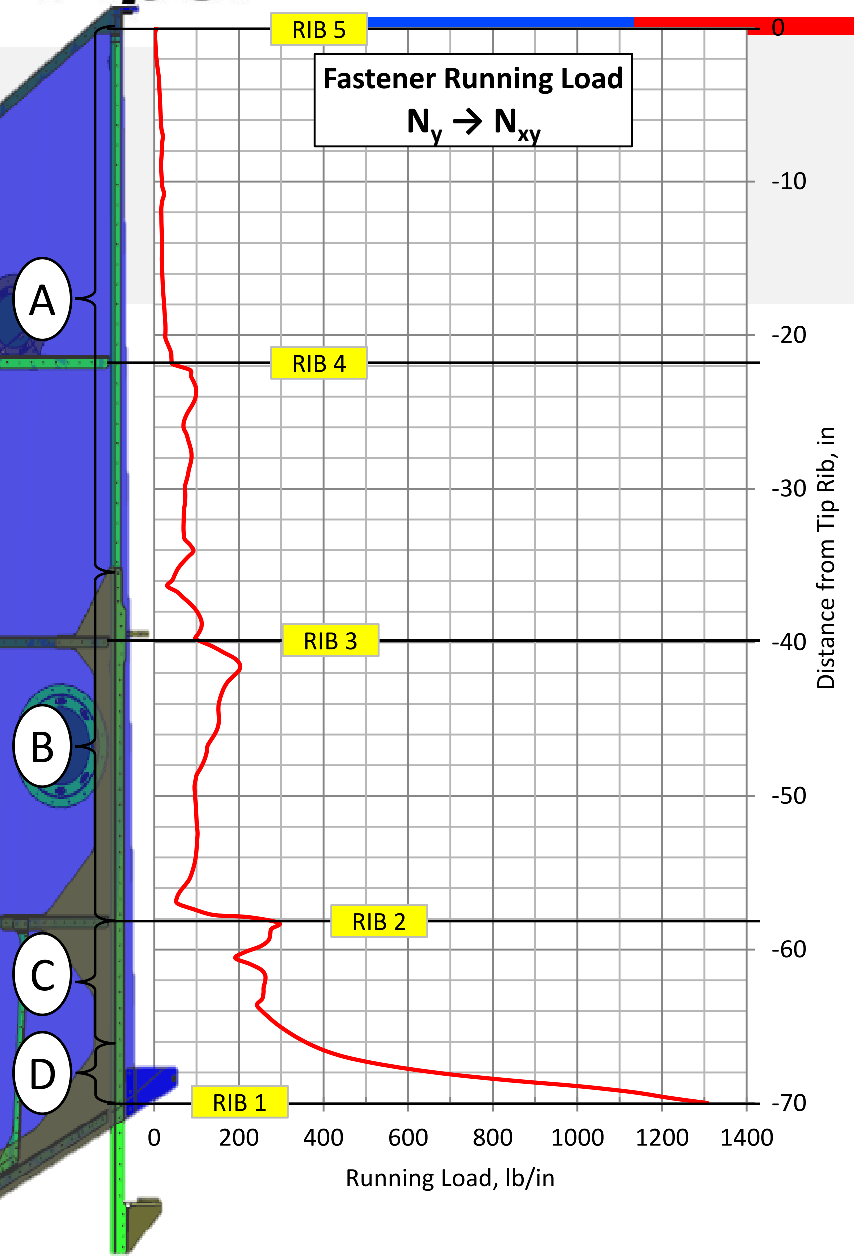




JOINT ANALYSIS USING VALIDATED NL FEM



- NL-FEM can disclose structural strength and stability, ***but does not demonstrate joint margins.***
- Fastener loads may revise skin thickness, rivet pitch, and rivet type/size.
- Inter-rivet buckling may revise attachment skin thicknesses and spacing.
- "Tear out" may revise attachment skin thicknesses and spacing.
- "Pull-Thru" may revise attachments, skin thicknesses and spacing.



Fastener shear – rear spar

A. YAA4 S = 1.000” t = 0.032” → P_{all} = 250 lb
$(N_y \rightarrow N_{xy})_{max}$ = 99 lb/in

B. YAA4 S = 1.000” t = 0.052” → P_{all} = 350 lb
$(N_y \rightarrow N_{xy})_{max}$ = 223 lb/in

C. ATZ4 S = 0.750” t = 0.052” → P_{all} = 558 lb
$(N_y \rightarrow N_{xy})_{max}$ = 518 lb/in

D. ATZ5 S = 0.644” t = 0.052” → P_{all} = 854 lb
$(N_y \rightarrow N_{xy})_{max}$ = 1307 lb/in

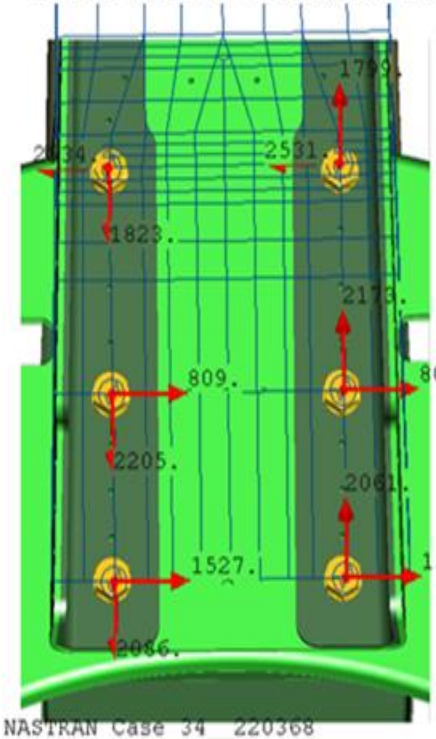
{Total load over 4 ATZ5’s = 2622 lb, or 656 lb/rivet}

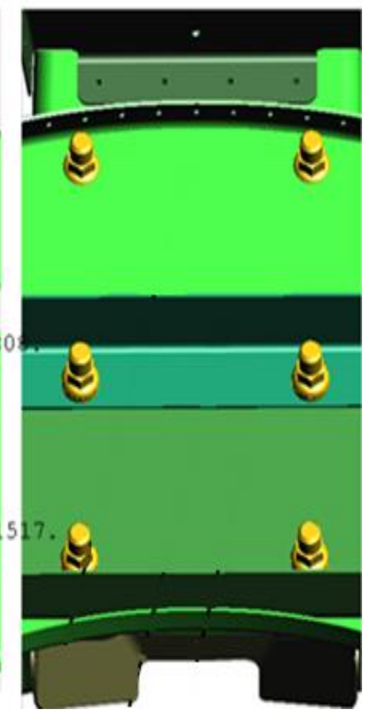
Note: discreet running load determinations for joints can be determined with the refined NL model and include skin buckling effects. Transverse (pull-thru loads) can be much more realistic than classical text would require.

Rear spar to fuselage joint

VT connections to fuselage structure

Rear spar with critical freebody loads, AN6 bolts





REAR SPAR	MAX	3121	LIMIT		AN6-7A	Ps_all =	8280	lbs
		4682	ULT	D =	0.375	Pt_all =	10100	lbs
		1.15	FF					

UPPER FASTENERS

Element 10100			Element 8657		
Output Set		3121	Output Set		3105
220368	2534	Bush X Force	220368	2531	Bush X Force
	-1823	Bush Y Force		1799	Bush Y Force
	-221	Bush Z Force		205	Bush Z Force

MID FASTENERS

Element 10101			Element 8658		
Output Set		2348	Output Set		2319
220368	-809	Bush X Force	220368	-808	Bush X Force
	-2205	Bush Y Force		2173	Bush Y Force
	329	Bush Z Force		-325	Bush Z Force

LOWER FASTENERS

Element 10102			Element 8659		
Output Set		2585	Output Set		2559
220368	-1527	Bush X Force	220368	-1517	Bush X Force
	-2086	Bush Y Force		2061	Bush Y Force
	37	Bush Z Force		-38	Bush Z Force

Minimum bearing margin - Spar

P max = 3121 lb lim

SPAR	t	Fbru e/D = 2.0		fbrg ult		MS_brg_ult Fbru/(FF * fbrg) -1 =
spar	0.100	119000	psi			
angle	0.100	119000	psi			
(combined)	0.200	119000	psi	62424	psi	

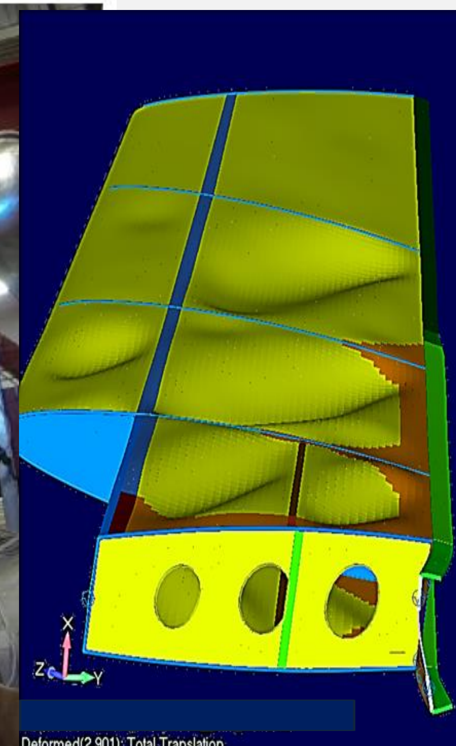
Minimum bearing margin – Fuselage structure

P max = 2585 lb lim

FRAME, CANTED, FS 298.895

84206	frame	0.050	121000	psi		
84207	dblr	0.050	121000	psi		
	(combined)	0.100	121000	psi	103406	psi

- Vertical –redesign resists critical design ultimate load



Vertical Stabilizer at load (load level shown is approximate, actual load held was calculated post-test using recorded data)

PA-46-600TP
Complex Fuselage Considerations

- Longeron Type Fuselage (Bruhn section C11.35)
  - Textbook panel compression + DT analysis limitation : per Bruhn section C11.36 para. 2. (no such thing as pure shear)
  - Non-circular (not covered by textbook theory)
  - Multiple openings (windshields, windows, doors)
  - Tail-cone and windshield transitions
  - Pressurization + Primary bending (not independent)
  - Discreet loads from wing, tail and seats
  - Stress limitations from thin sheet bending-permanent buckling
- Testing Required for Validation
- Buckled Shape Correlation
- NL-FEM of Revised Configuration
- Joint Analysis Using validated NL-FEM Results

Legacy Longeron Type Fuselage (Bruhn section C11.35)

- Paragraph C11.4 limitation

Some typical types of longeron structural systems cross-sections for a fuselage are shown in Fig. C11.44.

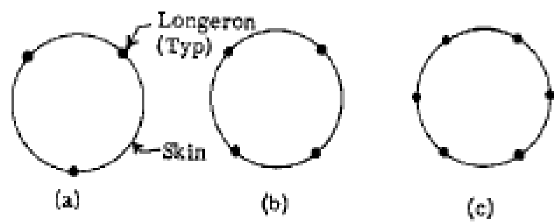


Fig. C11.44

- Paragraph C11.36. conservatism

2. Next determine the critical shear buckling stresses in the skin panels. Since compression stresses are nearly always also present in practical situations, pure shear buckling does not occur. Thus, as discussed in the case for stringer design, some rational interaction must be used to obtain a "reduced" shear buckling stress. This can be done, for example, by using some "average" compression stress in the panel, weighted toward the high side for conservatism. Thereby the interaction method of Article C11.32 can be used where

$$R_c = \frac{-\frac{B}{A} + \sqrt{(\frac{B}{A})^2 + 4}}{2} \text{, as in (74)}$$

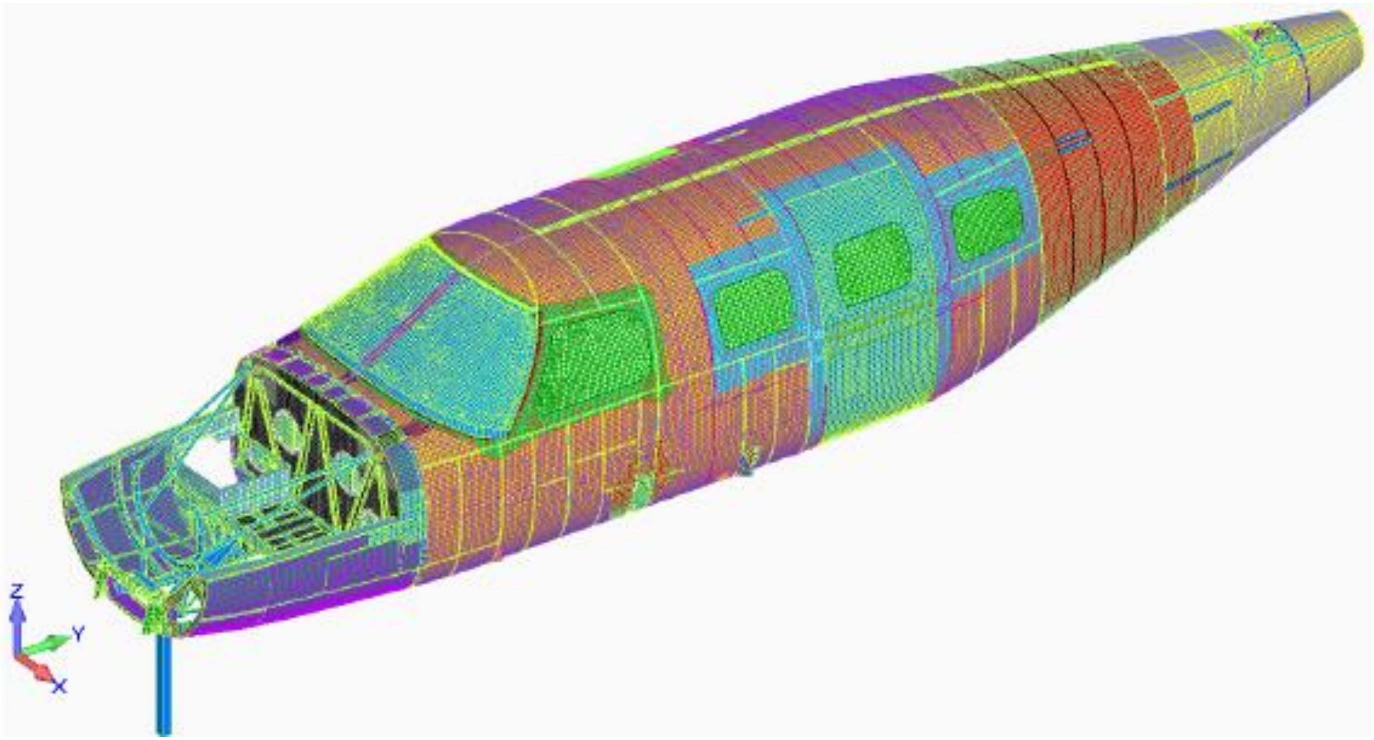
and (A) is determined for a curved panel of length "d" between rings and height (h) between longerons measured along the circumference as in Fig. C11.34. (B) is the ratio of the compression stress to the shear stress, (f_c/f_s), for the particular loading condition being investigated. The compression stress should be calculated as if the panel being calculated had not yet buckled. Then, as in equation (75)

$$f_{s_{cr}} = R_c F_{s_{cr}}$$

gives the reduced shear buckling stress, $f_{s_{cr}}$.

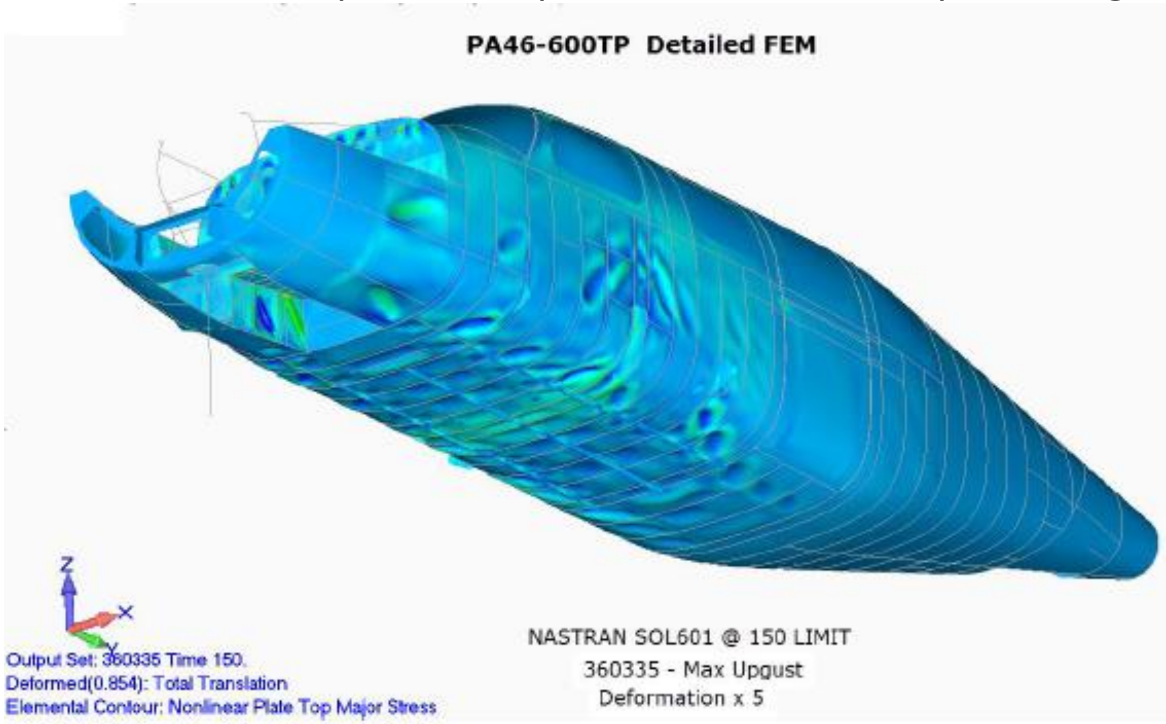
Fuselage NL stability gross stress model

- 38,622 nodes, 41158 elements
- Includes representative tail-cone to pressure dome
- Used for all required analyses for tested and non-tested conditions



NL FEM stability gross stress model solution

- Required for accurate for internal load distribution FAR 23.301 (c)
- Free from Textbook limitations and conservatisms
- Valid for forensic analyses
- Material Non-linearity can be exploited for thin sheet yield margin



Buckled skin panel analyses

- More accurate (post-buckled) model based panel buckling stress

9.1.1.1.6. Reduced Shear Buckling Critical Stress

The ratios A and B are calculated and used to determine R_c which is used to scale the shear buckling critical stress, $F_{s,cr}$, to the reduced value, $f_{s,cr}$:

$$A = \frac{F_{c_{cr}}}{F_{s_{cr}}}$$

$$B = \frac{f_c}{f_s}$$

Model based compression to shear ratio

$$R_c = \frac{-\frac{B}{A} + \sqrt{\left(\frac{B}{A}\right)^2 + 4}}{2}$$

$$f_{s_{cr}} = F_{s_{cr}} R_c$$

Accurate shear buckling stress in the in the presence of compression

For example, for panel 1000213, load case 1201 ultimate, without pressure:

$$A = \frac{1464\ psi}{2494\ psi} = 0.587$$

$$B = \frac{5966\ psi}{8347\ psi} = 0.715$$

$$R_c = \frac{-\frac{.715}{.587} + \sqrt{\left(\frac{.715}{.587}\right)^2 + 4}}{2} = 0.562$$

$$f_{s_{cr}} = 2494\ psi \cdot 0.562 = 1401\ psi$$

No permanent buckling < design limit load

- Low permanent buckling strength is a challenge and can drive a thickness and weight,

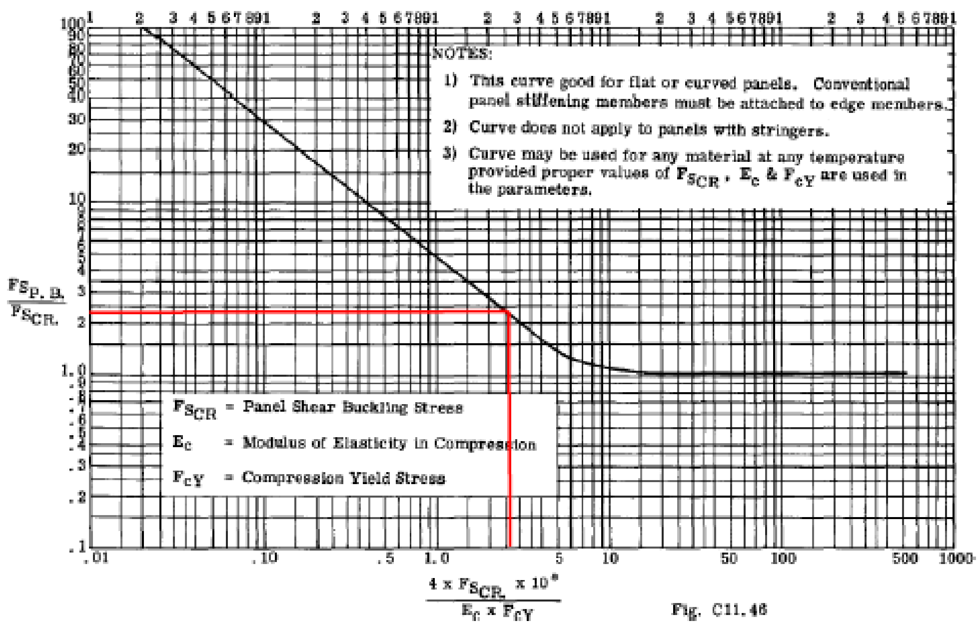


Fig. C11.46

- Alleviated with less conservative NL model determined combined fscr

Model derived nodal loads feed joint analyses with accurate (post-buckled) pull-off loads versus Bruhn design load (0.15 x t x Ftu) C11.24

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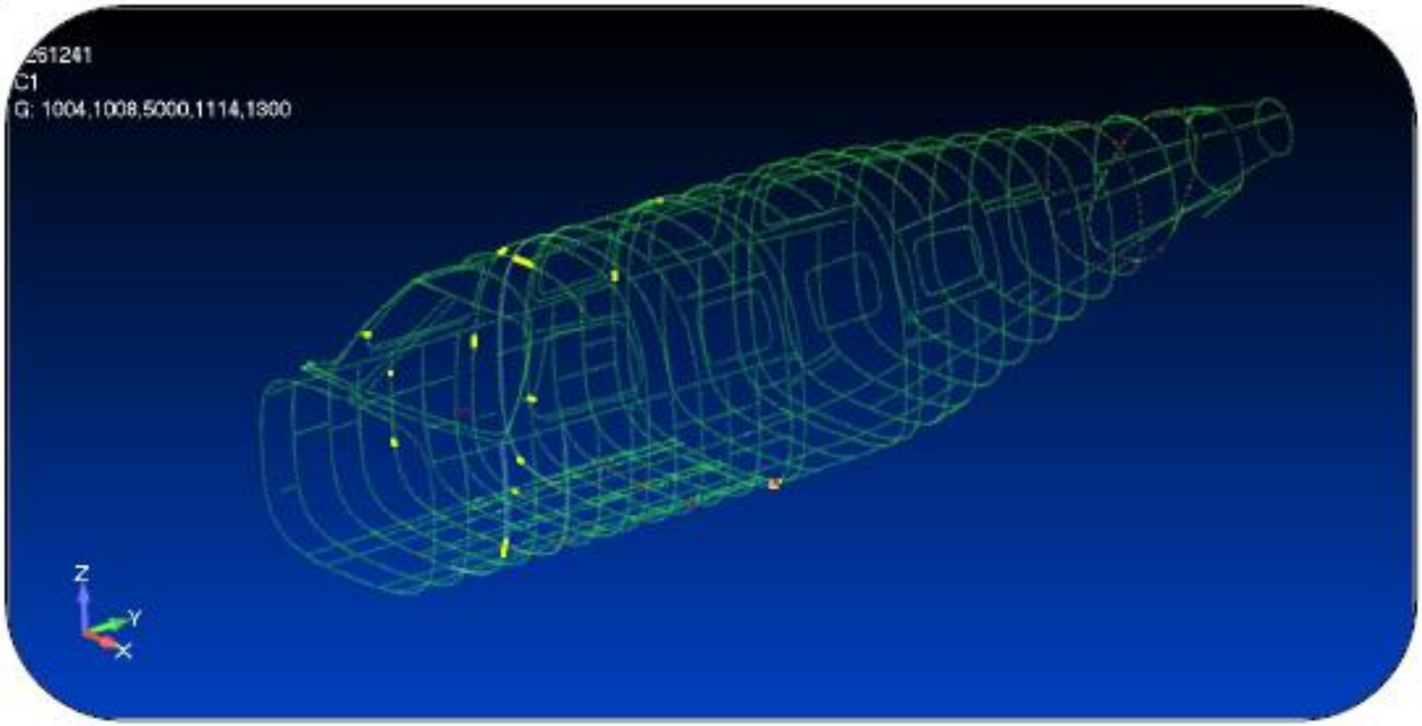
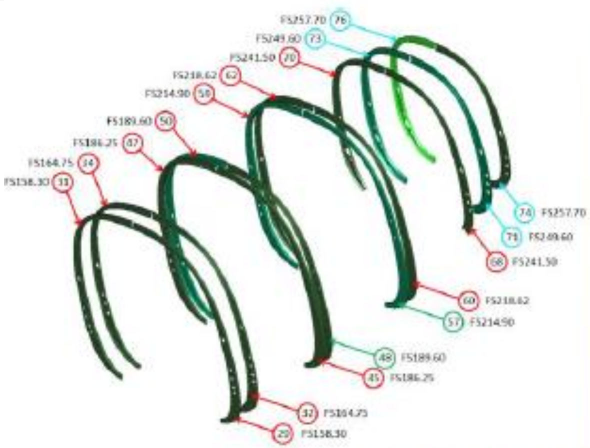


Figure 66 Higher Strength Fastener Replacement Locations Required by Stress

Leverage automated gross stress max/min load-case surveys for critical gross stress locations



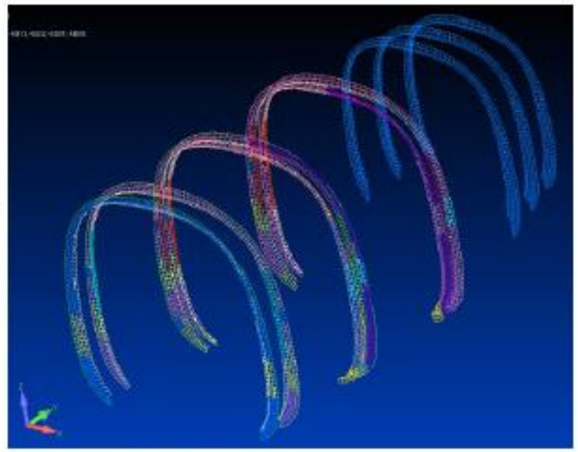


Figure 197 Side Fuselage Frames in Group 13,

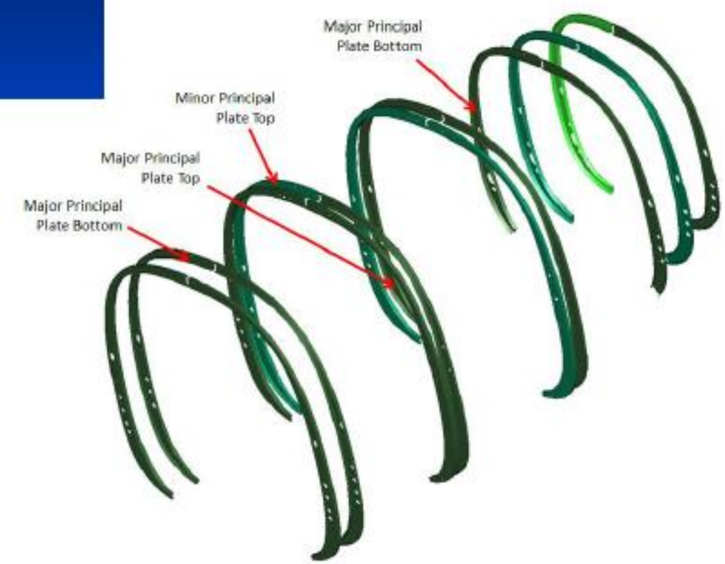
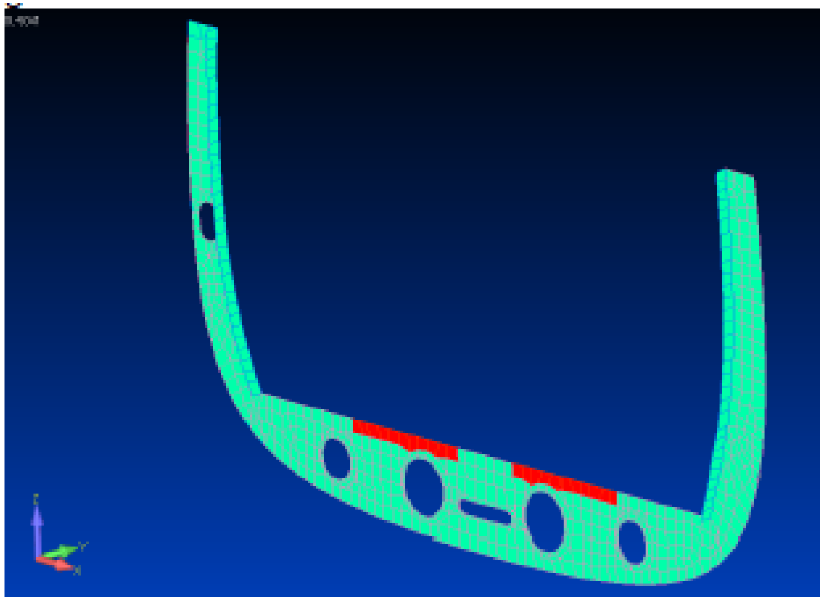


Figure 209 Peak ultimate principal stress locations, side frames

Model fidelity important for gross stress survey- peak model stresses may be conservative yet acceptable for cut-outs feature locations-or additional analyses may be required



82378, Frame FS233.87

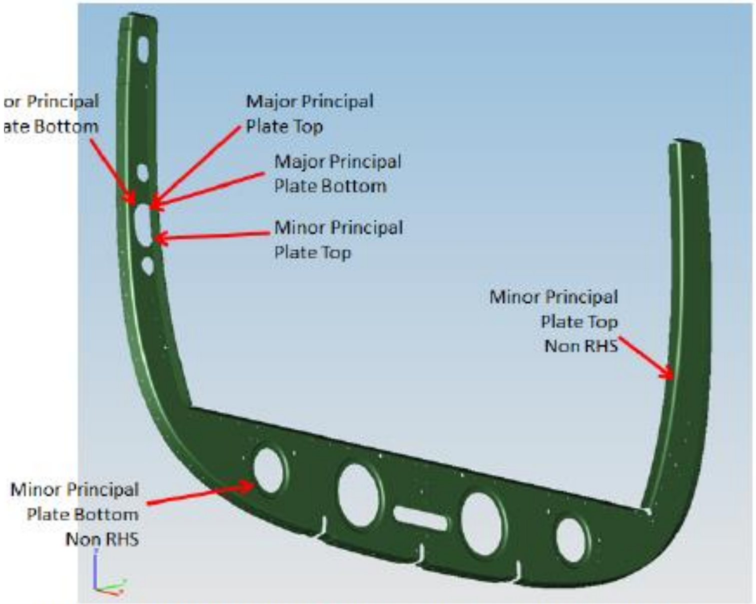


Figure 247 Peak ultimate principal stress locations, 82378

Complex Section analysis with accurate (post-buckled skin) free body loads

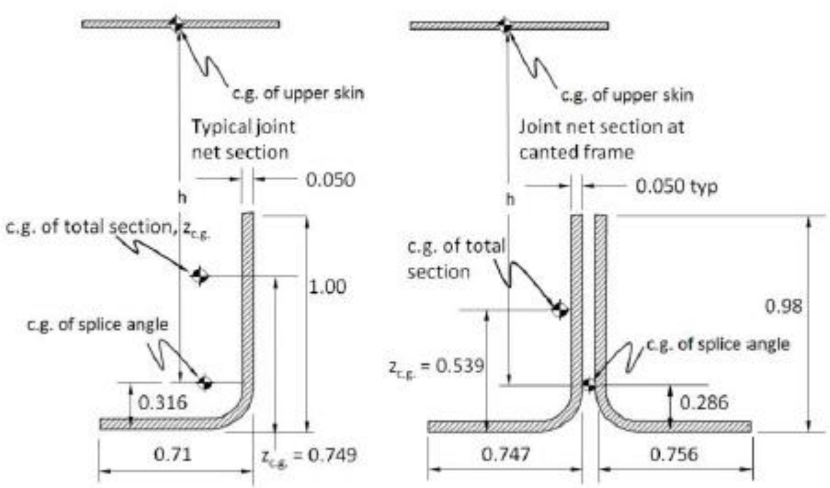
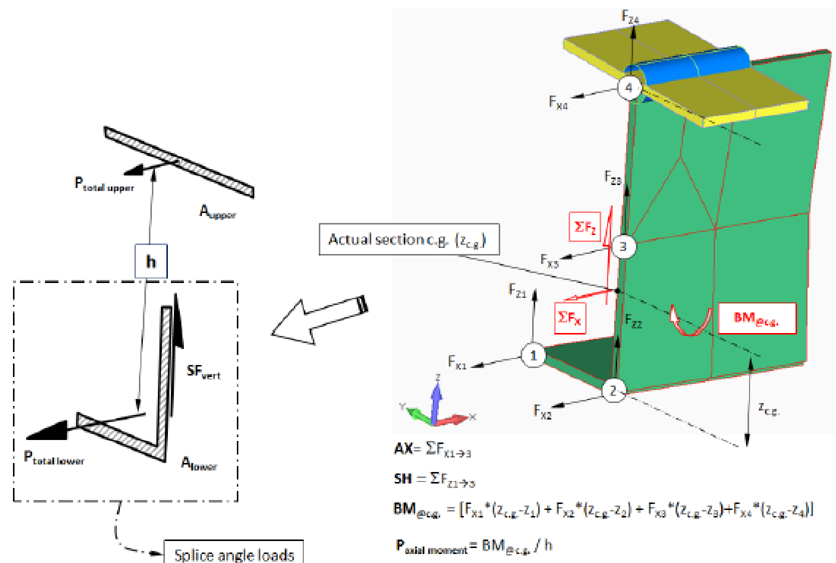


Figure 296 Upper Frame Joint Net Sections



$$AX = \Sigma F_{X1\rightarrow 3}$$

$$SH = \Sigma F_{Z1\rightarrow 3}$$

$$BM_{@c.g.} = [F_{X1}*(z_{c.g.}-z_1) + F_{X2}*(z_{c.g.}-z_2) + F_{X3}*(z_{c.g.}-z_3) + F_{X4}*(z_{c.g.}-z_4)]$$

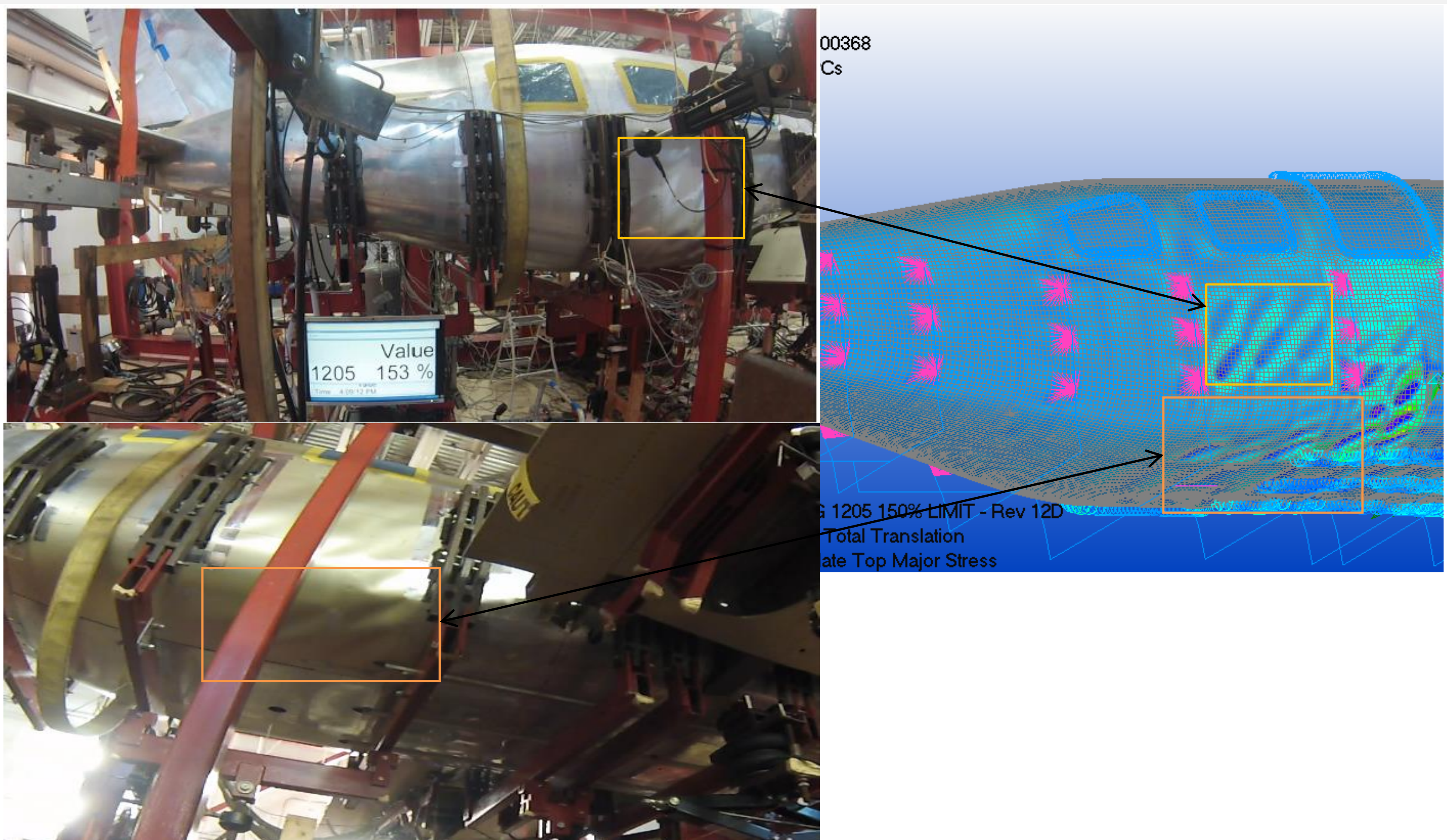
$$P_{axial\ moment} = BM_{@c.g.} / h$$

$$P_{total\ upper} = AX * A_{up} / (A_{up}+A_{low}) - BM_{@c.g.} / h$$

$$P_{total\ lower} = AX * A_{low} / (A_{up}+A_{low}) + BM_{@c.g.} / h$$

Figure 297 Transformation of Freebody Nodal Loads to Splice Joint Loads

Test versus Theory Aft fuselage



Test versus Theory: Forward Fuselage