Ogden Air Logistics Center

U.S. AIR FORCE

A-10 Modeling Methods to Support Damage Tolerance Analyses, Fleet Sustainment and Repairs

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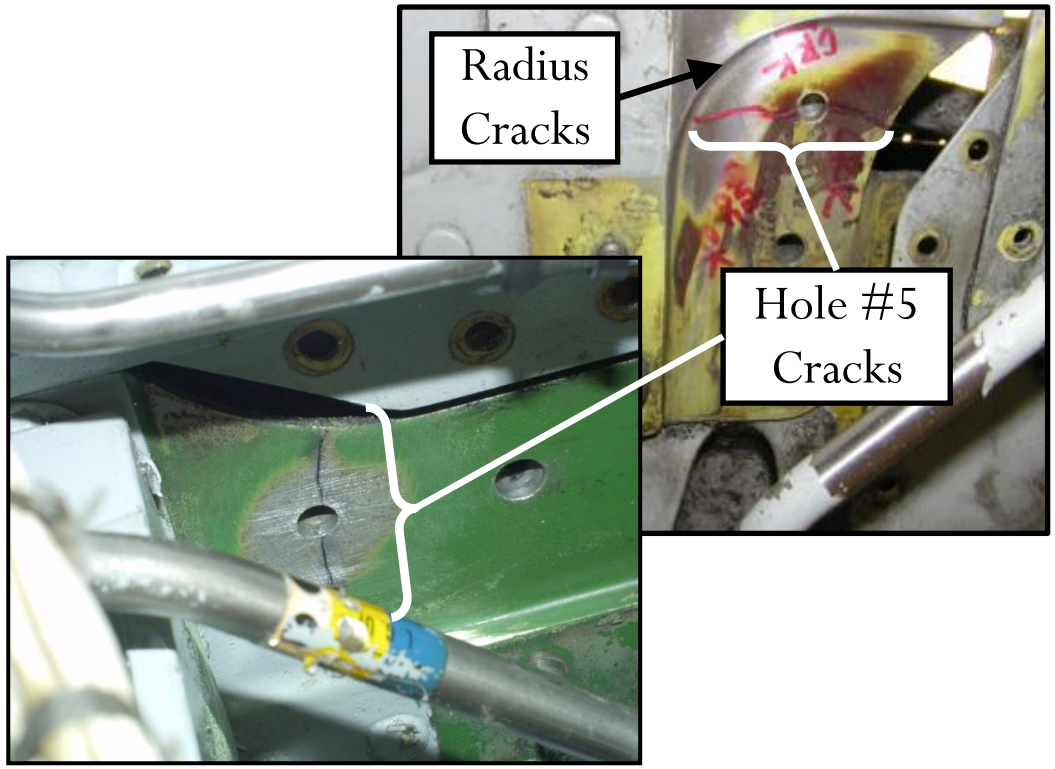
Background – Recent A-10 Cracking

- Two cracking situations:
  - Wing Cracking
    - Cracks discovered in A-10 lower wing skin in 2008.
  - Fuselage Cracking
    - Fuselage fatigue test upper longeron failure 2007.
    - Initiated inspection as a result on failure.
    - Significant crack discovered on aircraft in 2009.

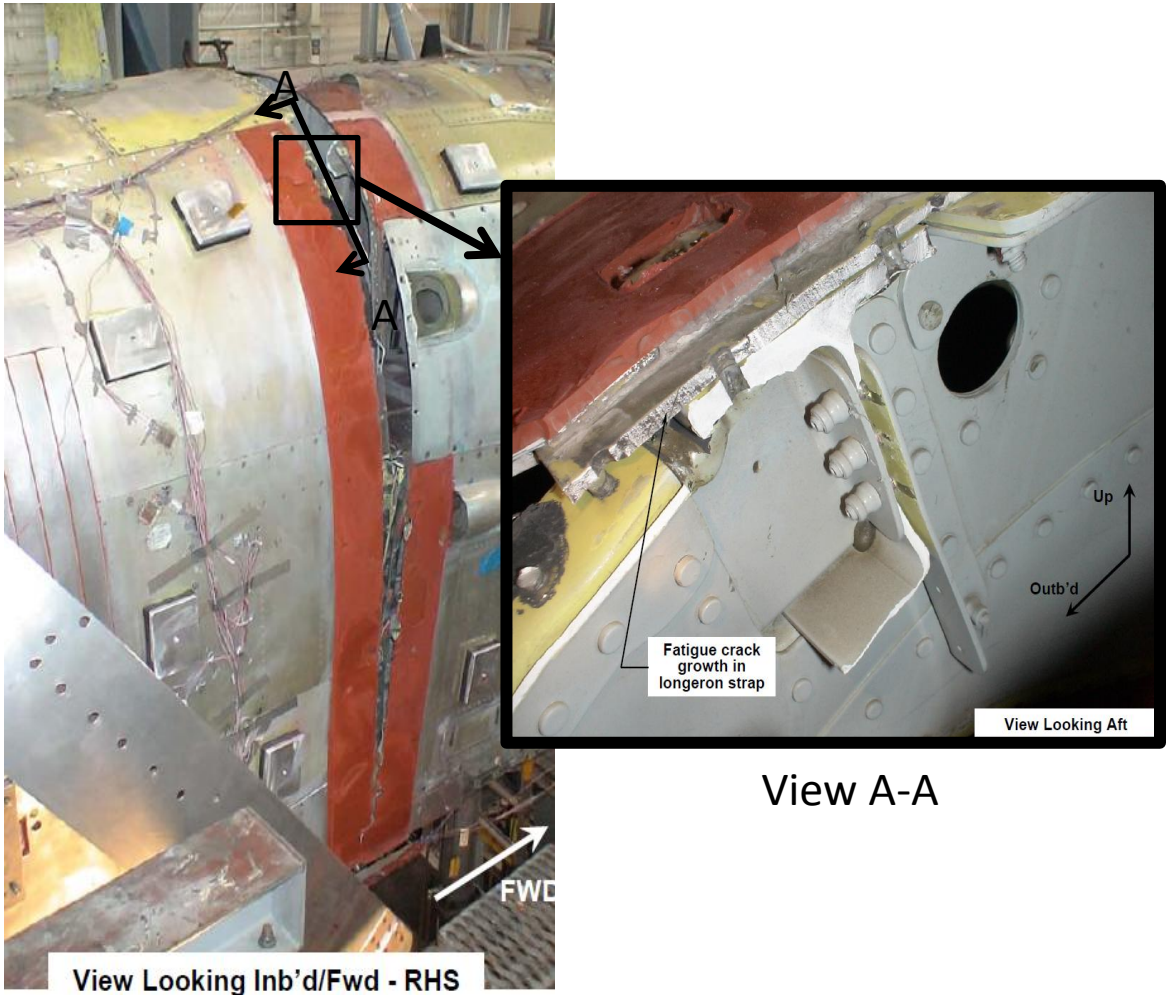
Lower Aft Wing Skin Cracking

- Increase occurrence of 202's
  - New inspection implemented
- First 6 Out of 6 Aircraft Inspected were Cracked Wing Cracking
- Significant Cracks Found in Two General Areas
  - Hole #5
  - Radius



Fuselage Cracking

- 2006 Fuselage Fatigue Test.
- 2007 Fuselage Failure due to severed Longeron.
- Longeron is made up of two components.
  - Longeron plate and longeron 'J' extrusion
- Cracking originated in upper longeron plate.
- Load redistribution in the J extrusion resulted in unanticipated failure.



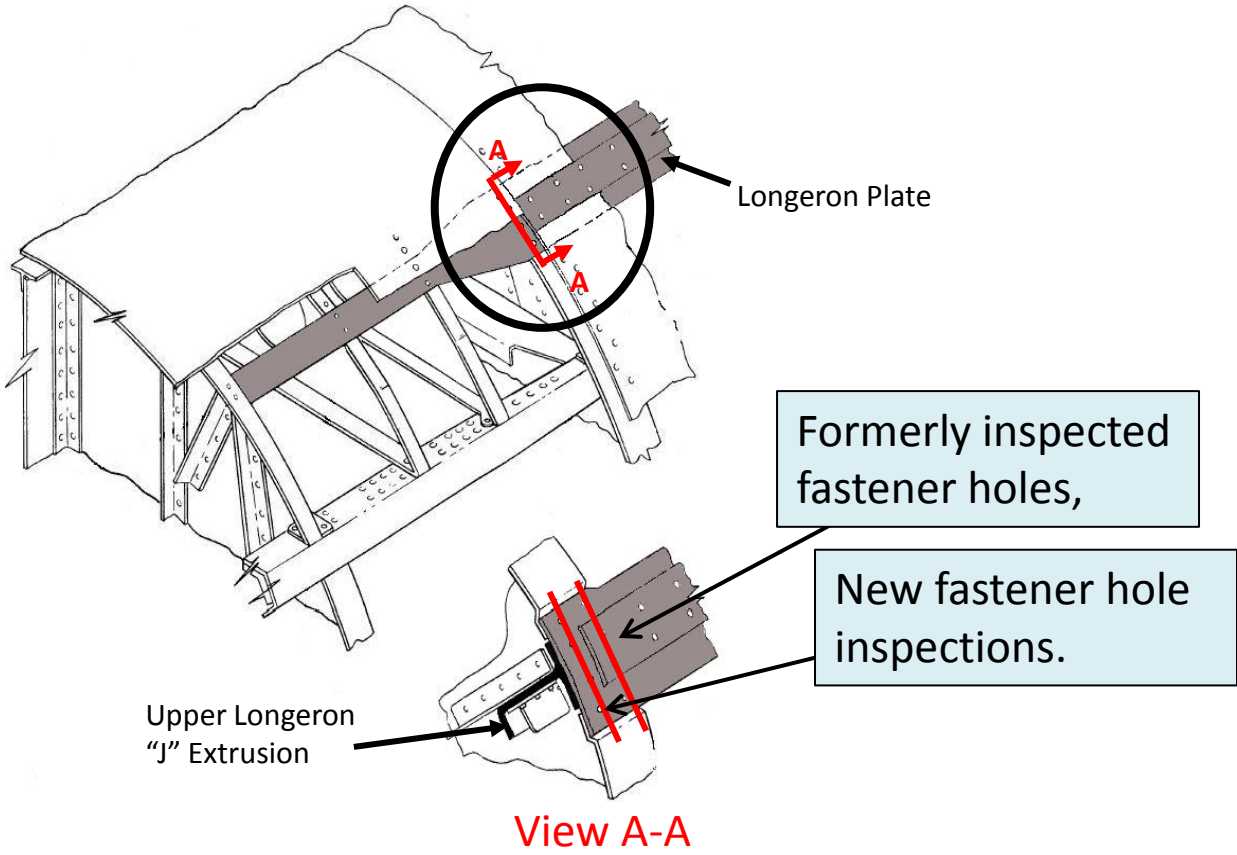
View A-A

Fleet Inspection Criteria

- Due to the catastrophic longeron failure a new inspection was generated for all A-10 aircraft in depot.

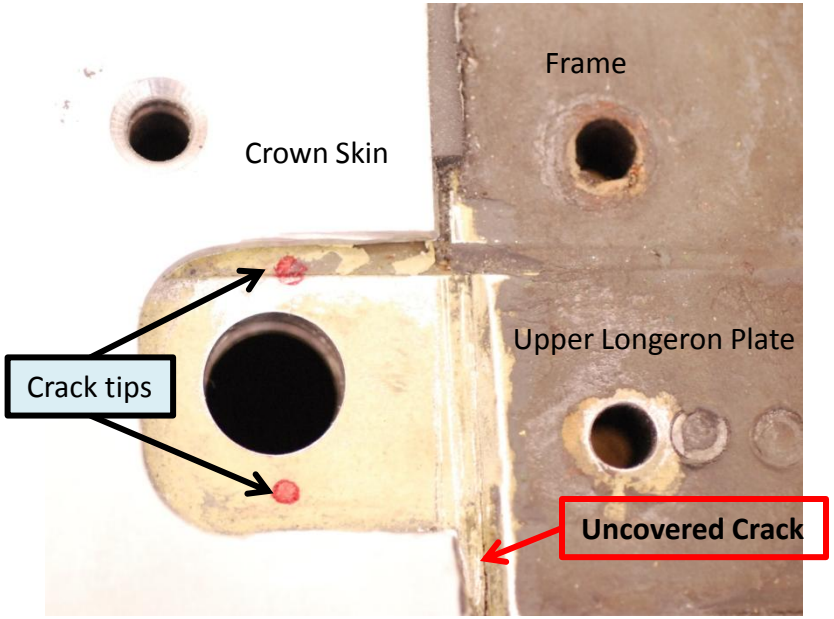
- Occasionally cracking indications were found at the formerly inspected fastener holes.

- Significant amount of cracking indications at fasteners holes were discovered in jets at the new inspected location.



First Critical Finding

- July 2009 a crack that was larger than the maximum oversize limit was discovered.
  - Crown skin was trimmed so the end of the crack could be seen.
- Along with a cracked fastener hole, a full radius crack was discovered just aft of the fastener hole.



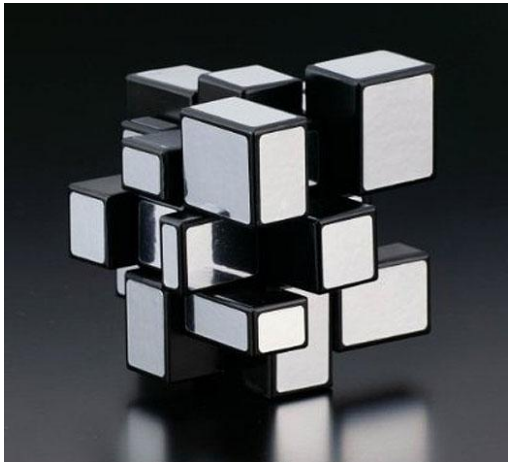
View of Oversized Fastener hole

Upper Longeron Cut Out

Back Side

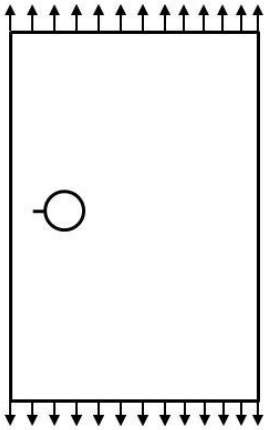
What are the Issues?

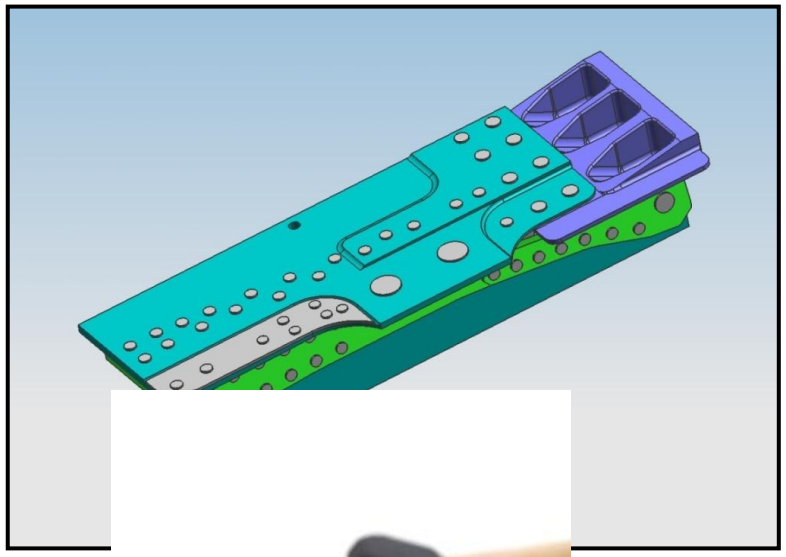
- Complicated Local Geometry
  - Thickness changes, radii, doublers, etc.
- Complicated Geometric Area
  - Splice between center and outer wing sections (Wing Only)
- Significant Load Transfer Between Components
- Unknowns
  - Local stresses, fastener loads
  - Load redistribution with cracks
  - Multiple cracking locations & sizes
  - Cause of the cracking
  - Limited fleet data



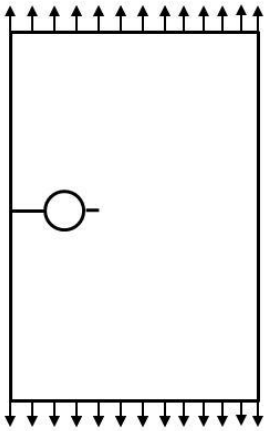
Classical Approach

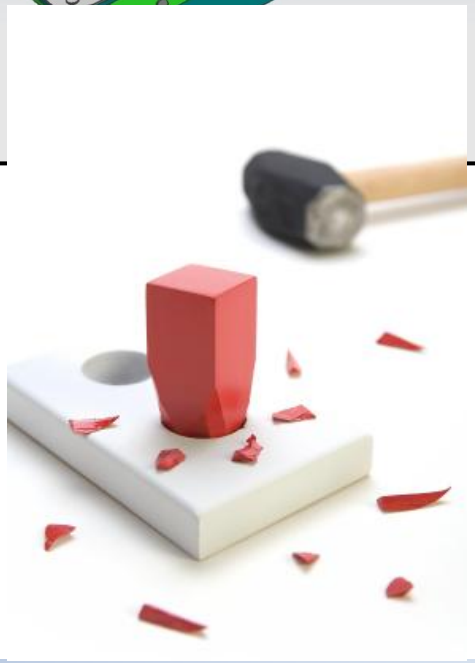
Classic Model





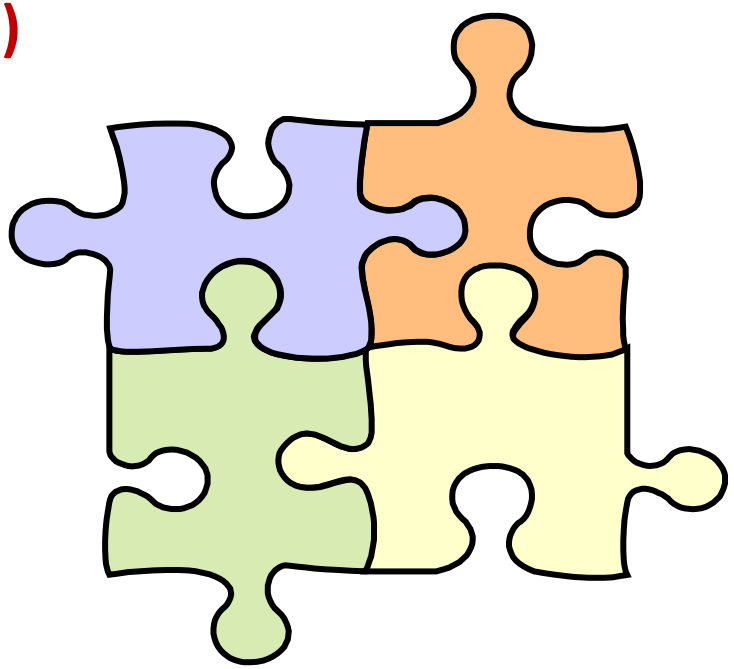
Classic Continuing Damage Model





Approach

- ASIP Process
  - Failure Analysis
  - Initial Damage Assessment
  - **Finite Element Modeling (FEM)**
    - **Global & Local FEM's**
  - **Crack Growth Analyses**
  - *Updated Risk Assessment*
  - *Field Inspections & Repairs*
  - *Long Term Repairs*



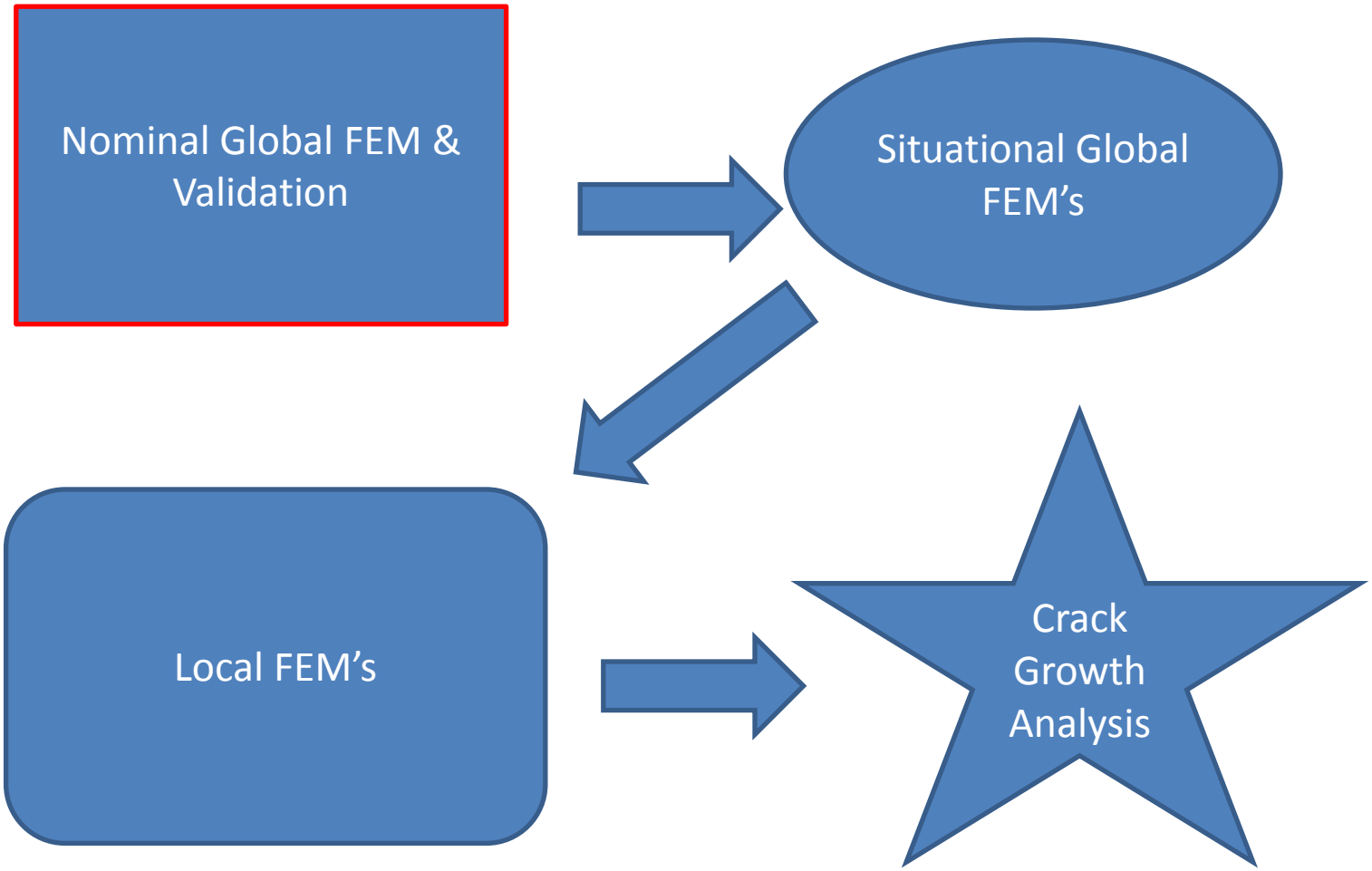
Global FEM Development

- 3D Finite Element Model (FEM) Developed
  - Advantages of the 3D model verses a 2D model
    - More Realistic
    - 3D cracks
    - Geometric detail
    - Redistributed stresses and Pin loads
    - Visual representation
  - Disadvantage of the 3D model
    - Long solve times

Global FEM Development

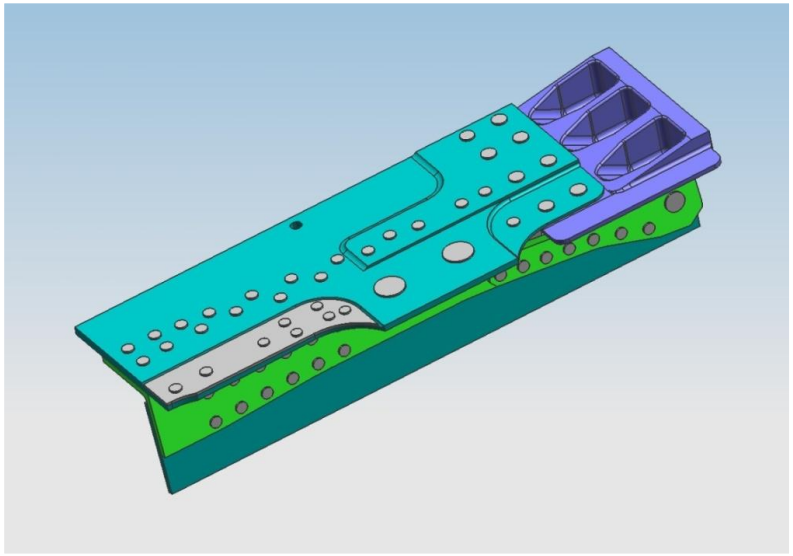
- Global Modeling and Simulation Approach
  - An accurate depiction of the structure
    - Adjacent constraints modeled
      - Frames/Bulkheads
      - Skins
    - Loading the structure with tensile loads and shear in skins
      - Loads allowed to be reacted in the model
    - Contact Surfaces
    - 3D Fasteners

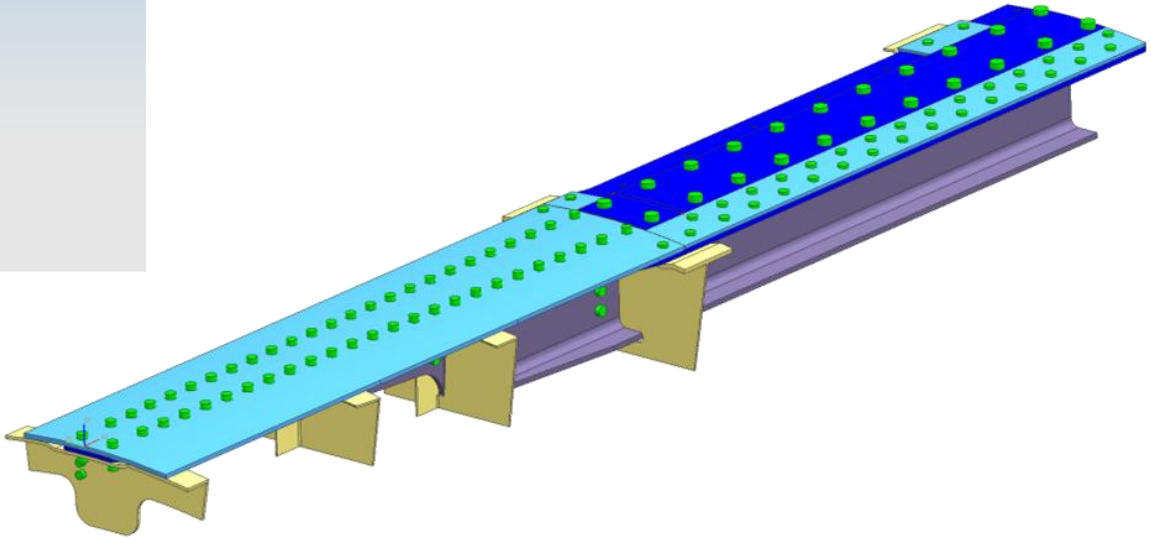
Global & Local FEM Flow Chart



Global Nominal FEM & Validation

- Nominal Wing Skin and Fuselage Models

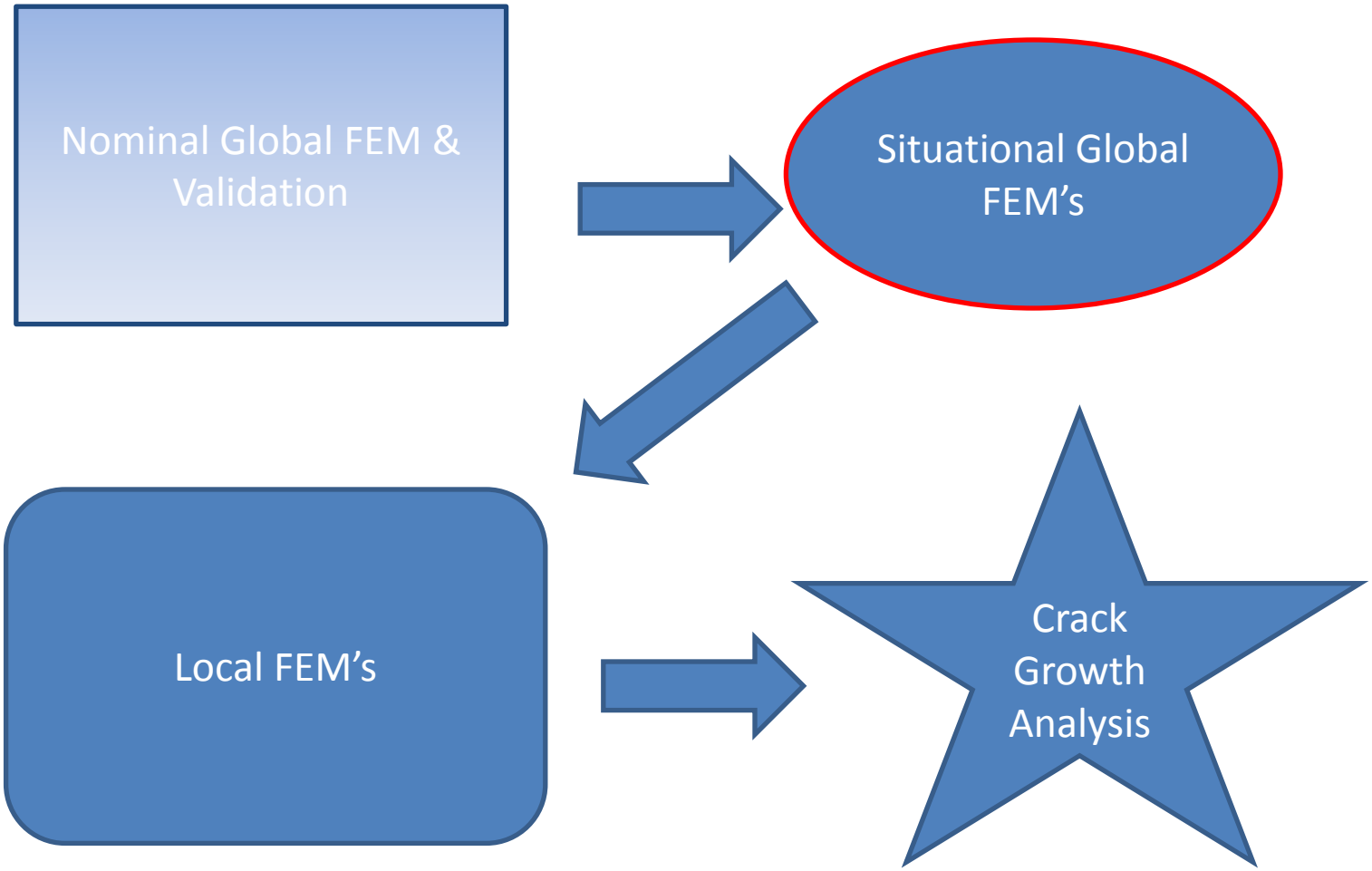




Global Nominal FEM & Validation

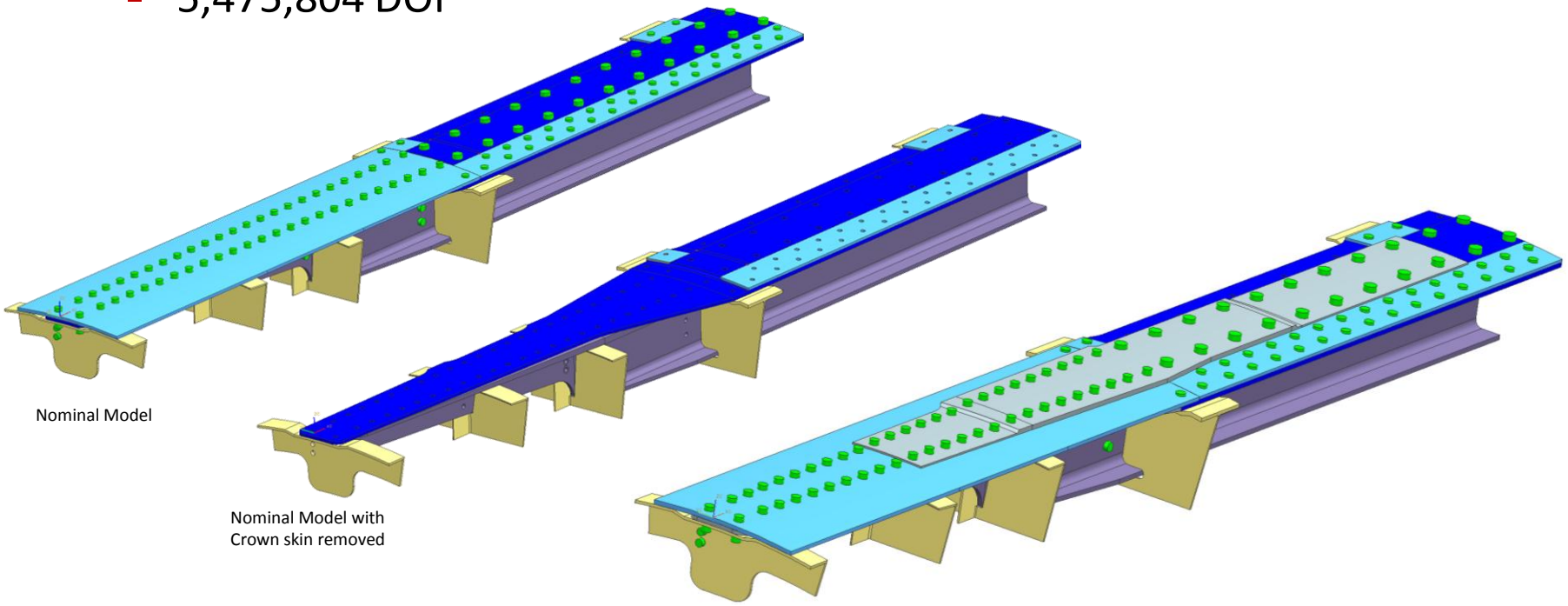
- Basic Process of Validating the Nominal Model
  - Loaded with limit loads
    - Stresses sampled at a strain gauge locations
  - Loads are scaled based on strain gauge data
    - Stresses are checked at multiple strain gauge locations
    - Tuned loads are applied to subsequent models
  - The validated loading is applied to other configurations of the model

Global & Local FEM Flow Chart



Situational Global FEM's

- Fuselage Nominal configured FEM.
  - 5,248,800 DOF
- Fuselage Repair Configuration
  - 5,475,804 DOF



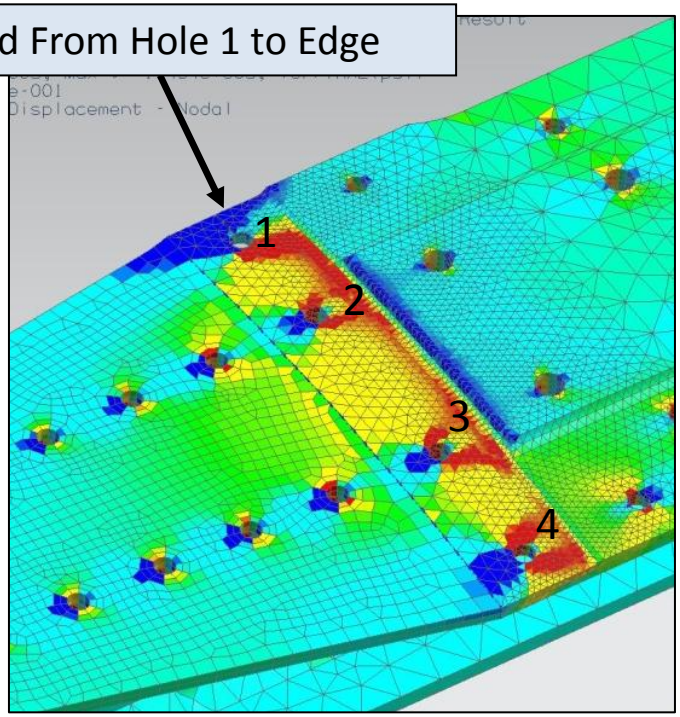
Nominal Model

Nominal Model with
Crown skin removed

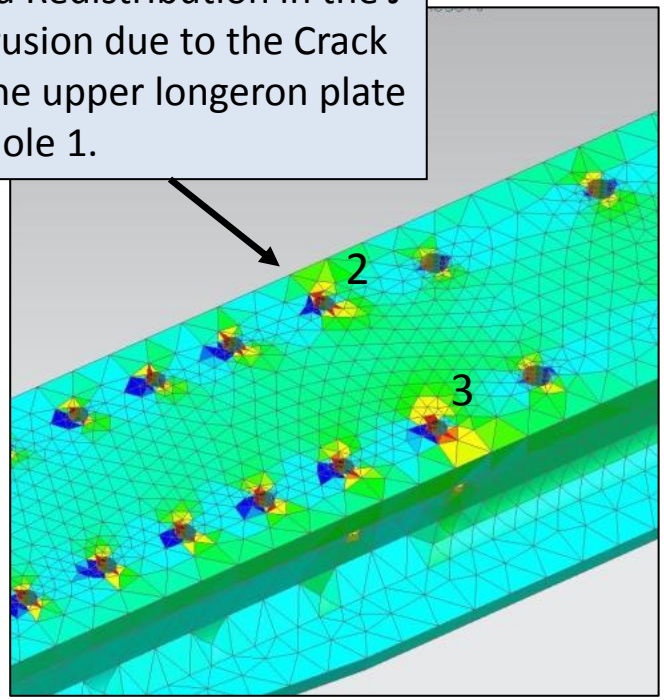
Repaired Model

Situational Global FEM's

Cracked From Hole 1 to Edge

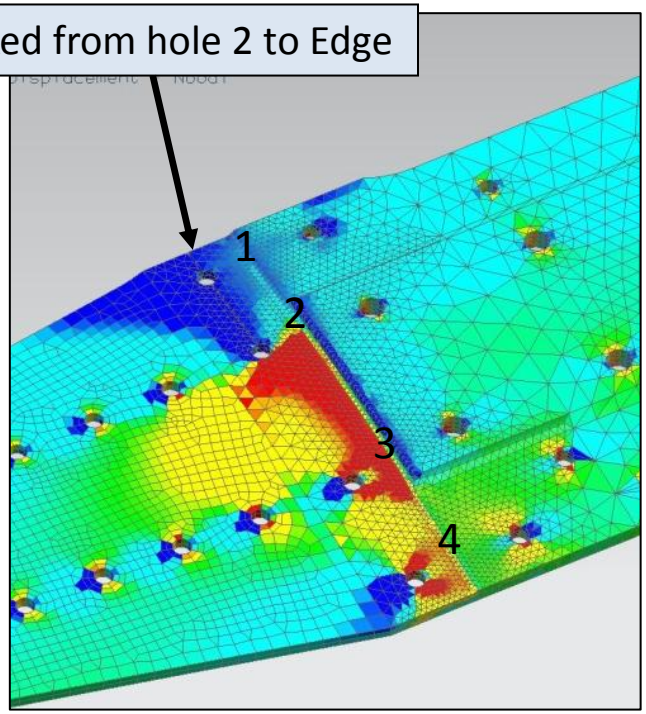


Load Redistribution in the J Extrusion due to the Crack in the upper longeron plate at Hole 1.

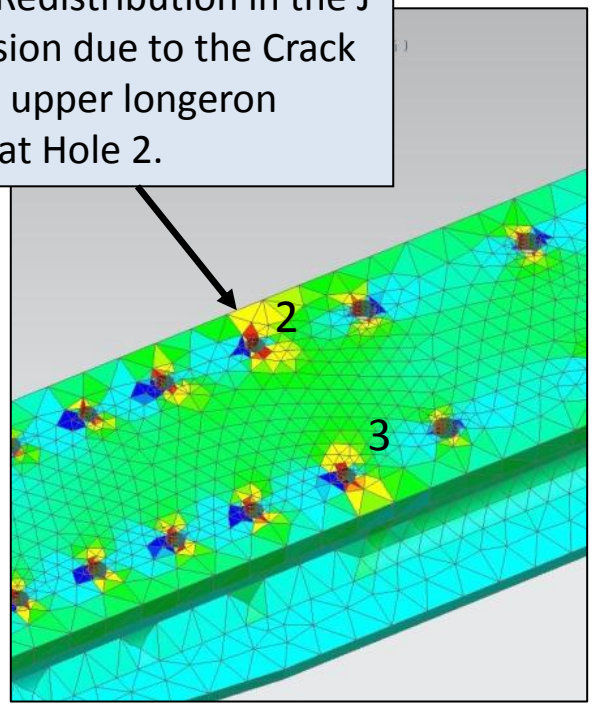


Situational Global FEM's

Cracked from hole 2 to Edge

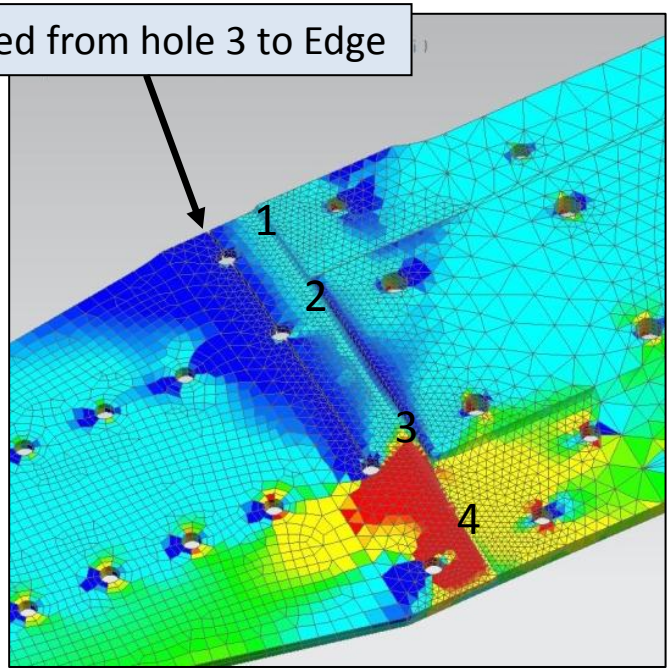


Load Redistribution in the J Extrusion due to the Crack in the upper longeron plate at Hole 2.

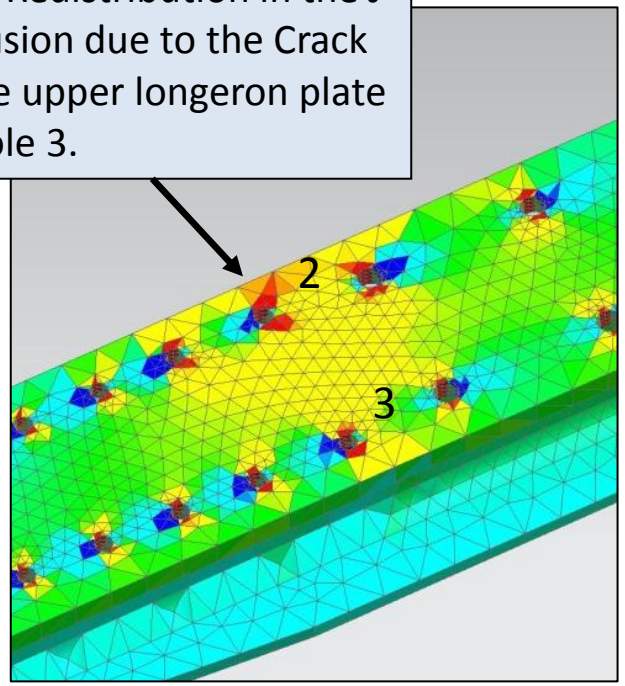


Situational Global FEM's

Cracked from hole 3 to Edge

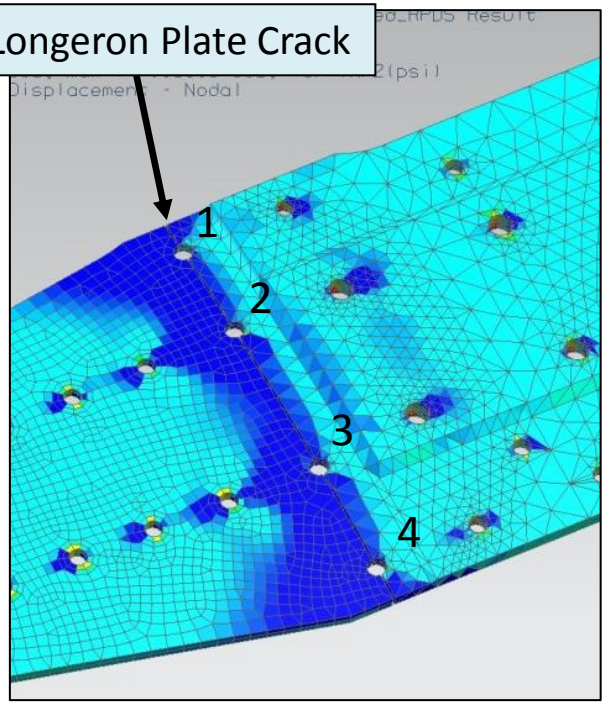


Load Redistribution in the J Extrusion due to the Crack in the upper longeron plate at Hole 3.

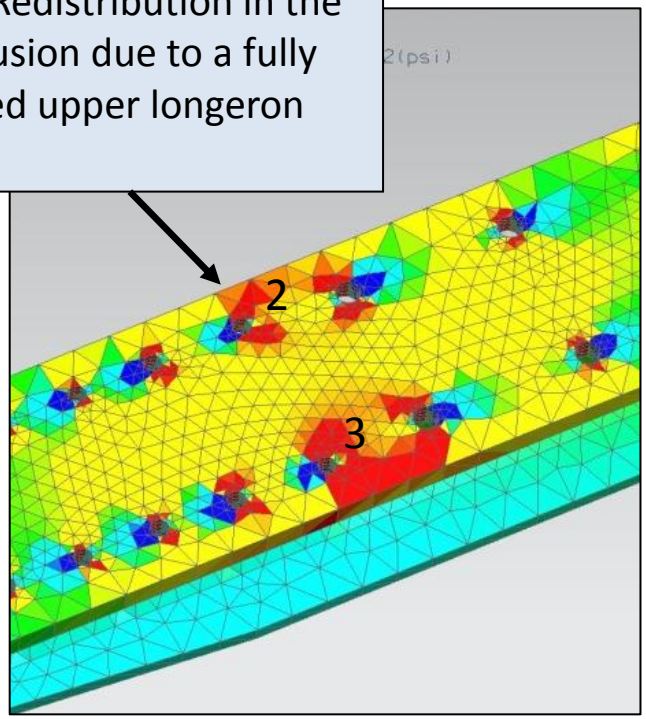


Situational Global FEM's

Full Longeron Plate Crack

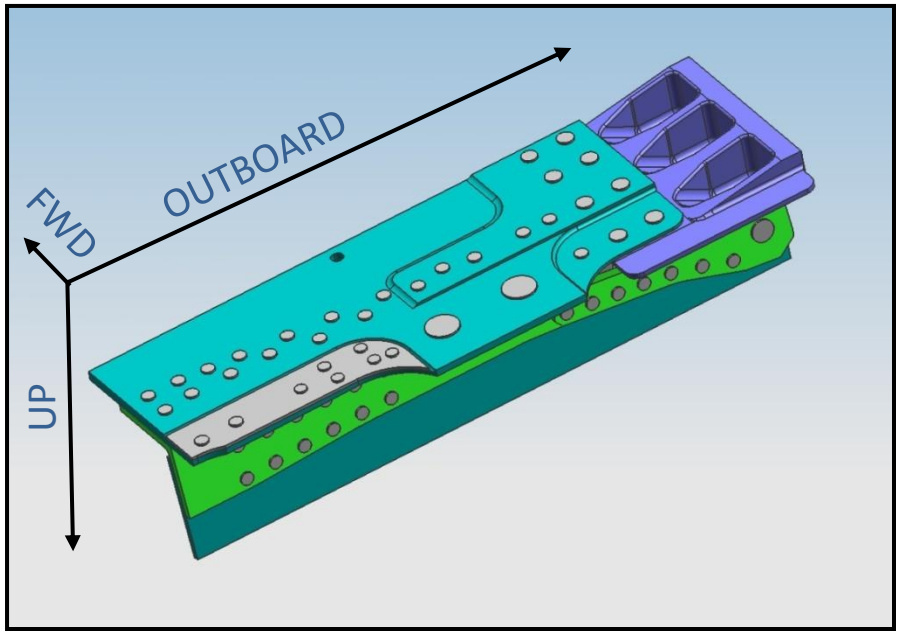


Load Redistribution in the J Extrusion due to a fully cracked upper longeron plate.

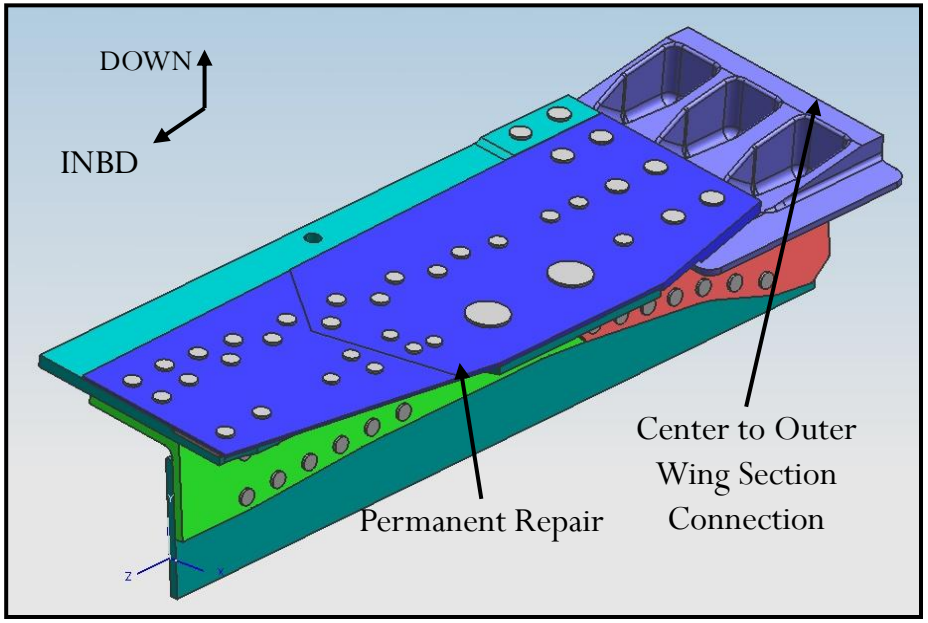


Situational Global FEM's

- Wing Skin Nominal & Repaired configured FEM's.

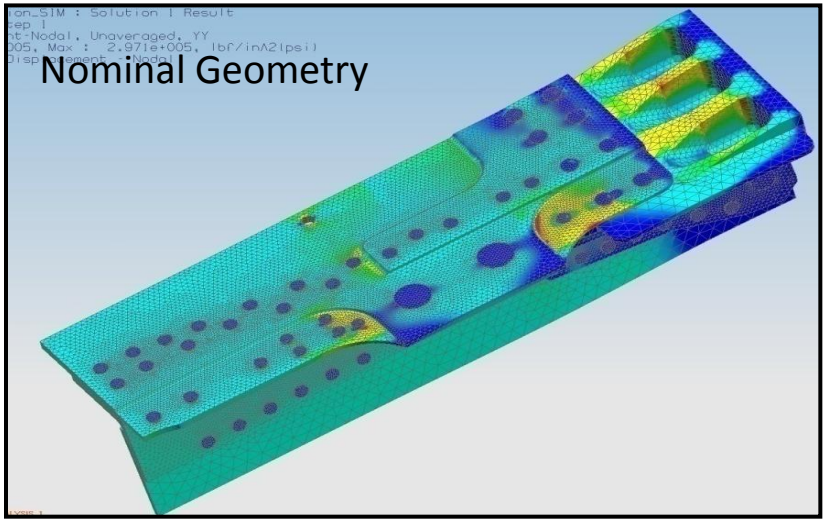


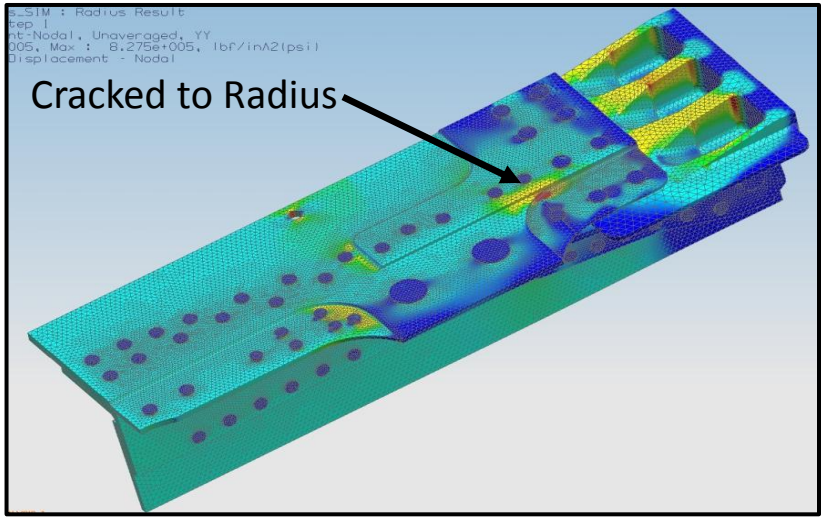
Nominal Model

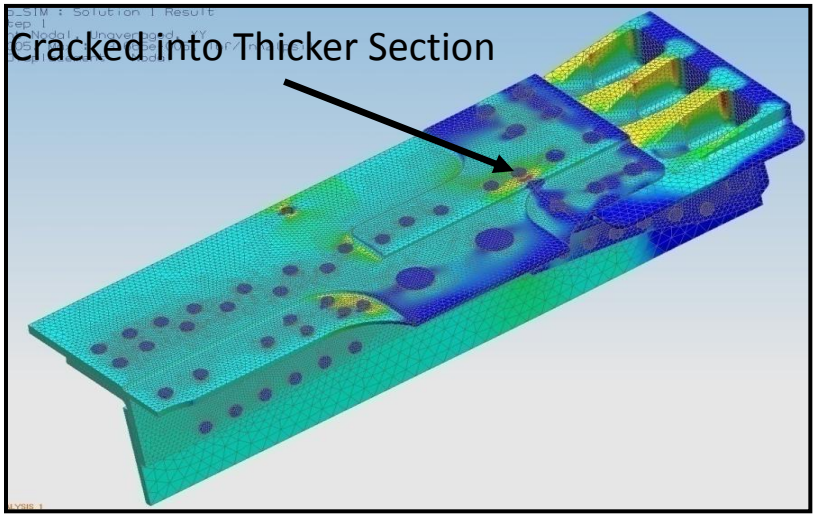


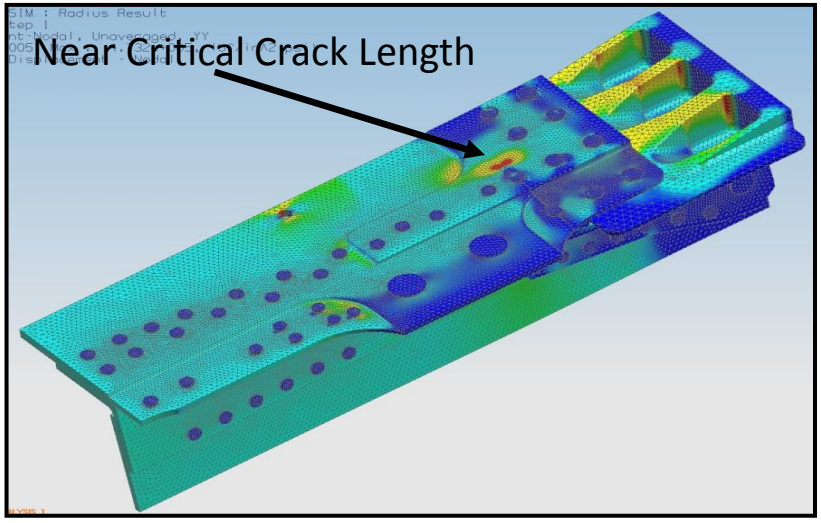
Repaired Model

Situational Global FEM's





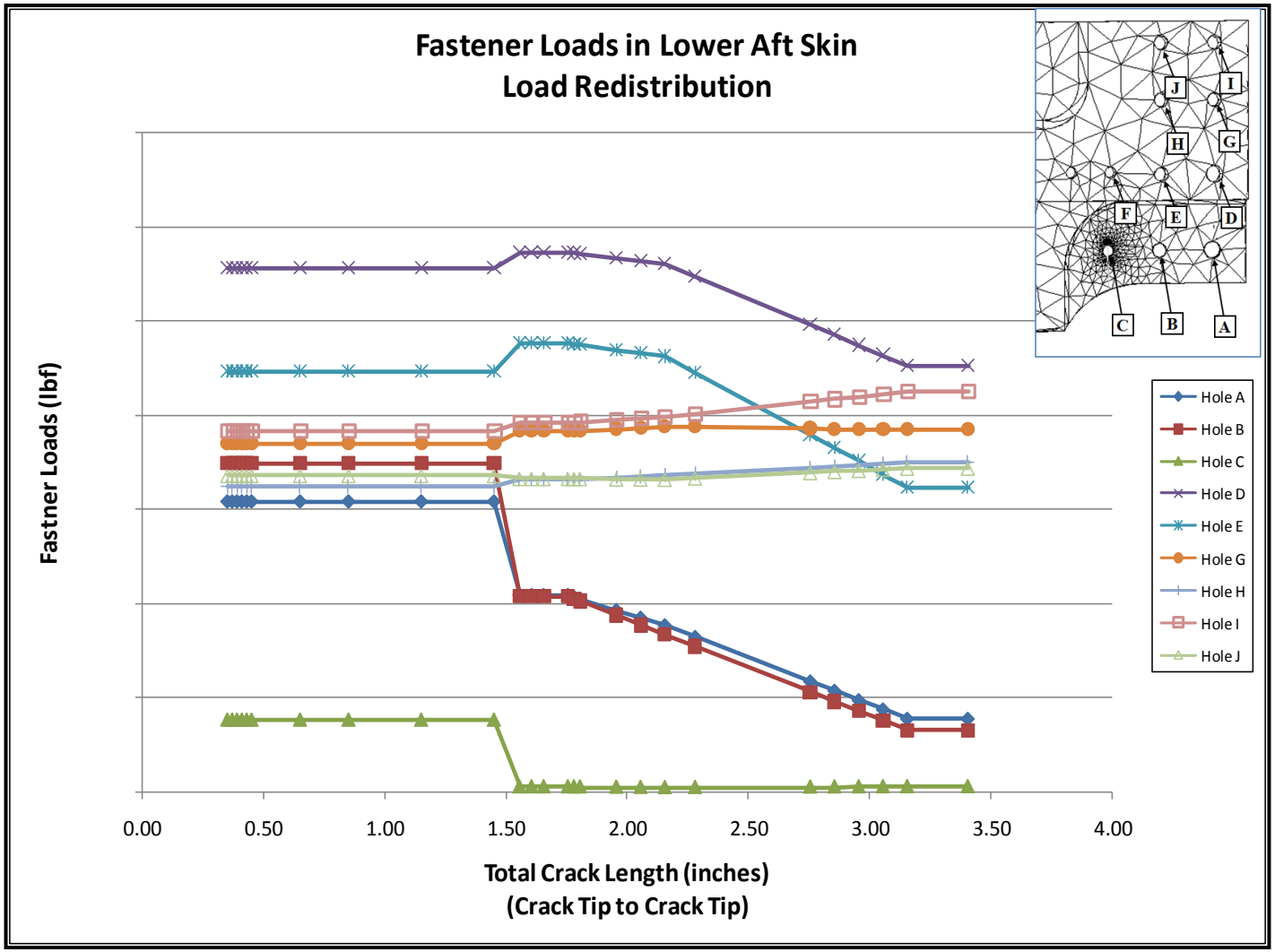




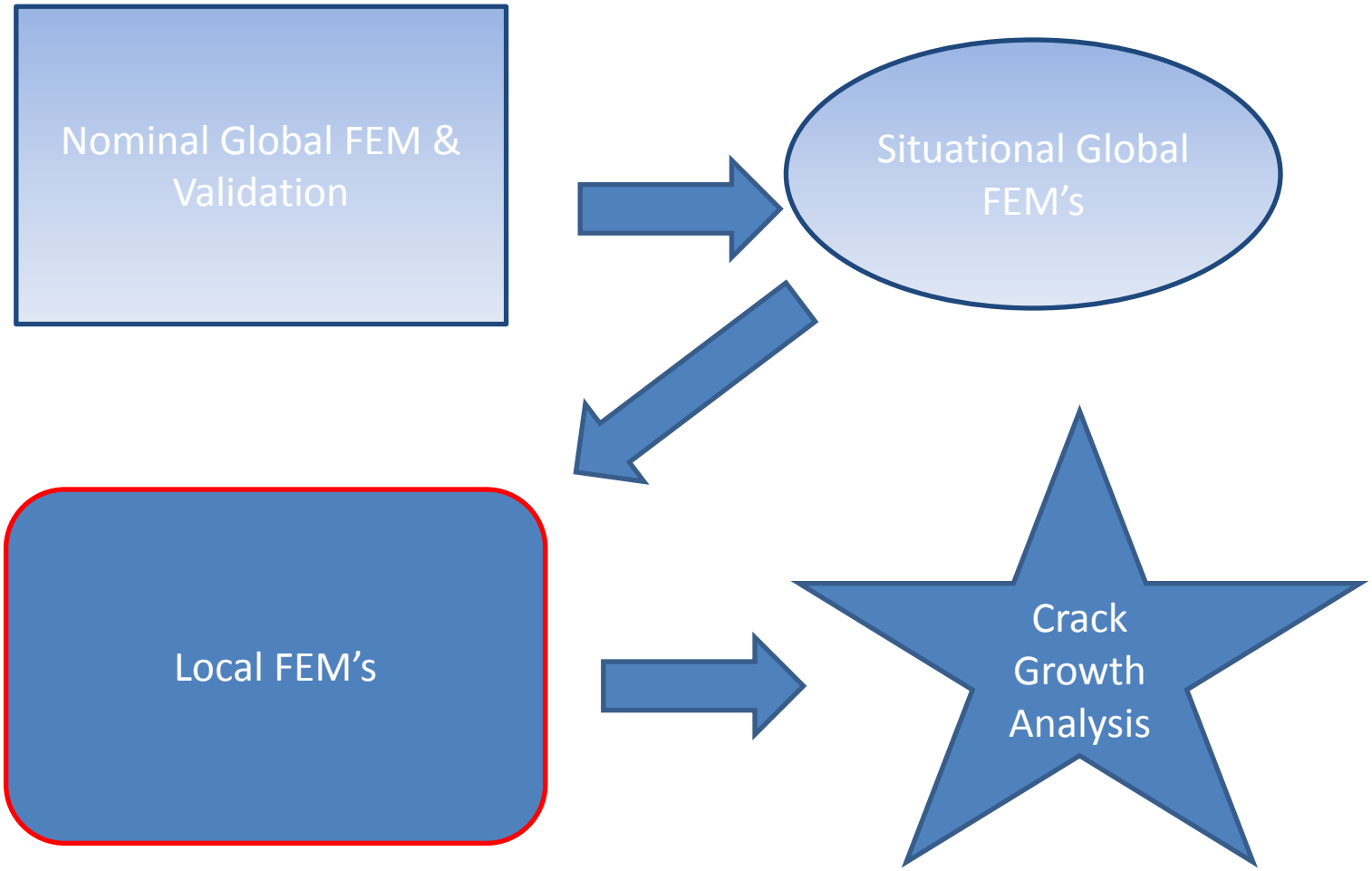
Global FEM Results

- Global FEM Outputs:
  - Pin loads
  - Redistributed Stresses
- Implications:
  - Data is easily available for current and future analysis.
  - Load redistribution can be quantified and used for local FEM, Damage Tolerance Analysis (DTA) and Risk Assessments.

Global FEM Results

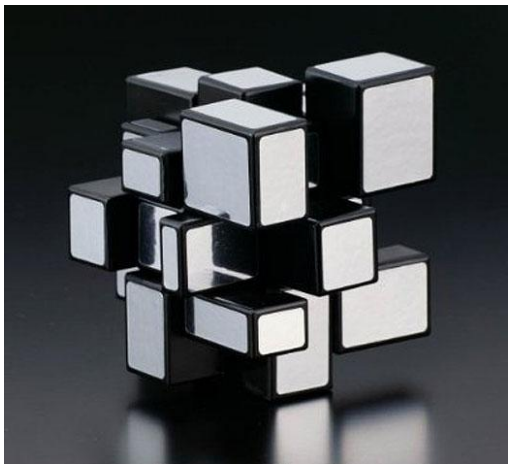


Global & Local FEM Flow Chart



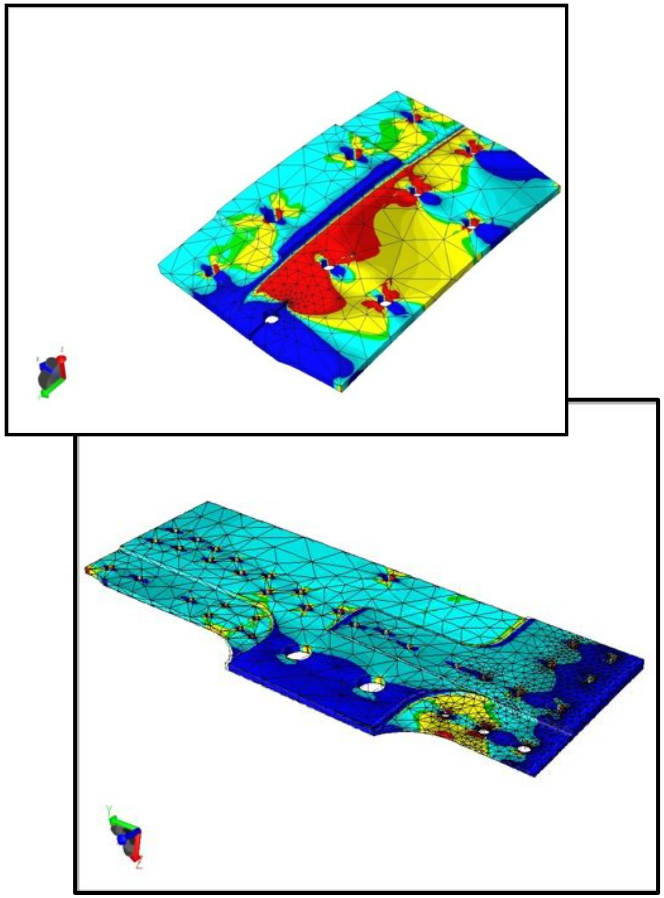
What are the Issues?

- Complicated Local Geometry
  - Thickness changes, radii, doublers, etc.
- Complicated Geometric Area
  - Splice between center and outer wing sections (Wing Only)
- Significant Load Transfer Between Components
- Unknowns
  - Local stresses, fastener loads
  - Load redistribution with cracks
  - Multiple cracking locations & sizes
  - Cause of the cracking
  - Limited fleet data



Local StressCheck (SC) FEMs

- Inputs
  - Fastener loads from Global FEM
  - Load redistribution as a result of cracks
  - Component geometry
  - Embedded cracks
- Outputs
  - Stress intensity as a function of crack length
  - Various cracking scenarios



Local StressCheck FEMs Approach

- Parasolid geometry imported into StressCheck
- Inputs parameterized for COM capability
- Cracks embedded in model

Start Solving on Row number	12	
Solution Options	Check & Solve	
Viewing Options	Show SC	
Save DB files (Yes/No):	Yes	
Auto Mesh	Yes	
Combine Output Files?	Yes	
Output Filename	CombinedFiles	
Image Capture (If yes include view name)	Yes	Stress_Image

Run

Input Data

			---------- Enter Parametric Parameter Names Below ----------										Solution ID	Extraction ID		
	Output File Name	p-level	CrackLength	stress	S1	S2	S3	S4	S7	S8	S9	S10				
cc_assymetric.sci	0050cc	4	0.050	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	KTop		
cc_assymetric.sci	0075cc	4	0.075	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	KTop		
cc_assymetric.sci	0100cc	4	0.100	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	KTop		
tc_assymmetric.sci	0125tc	4	0.125	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	K		
tc_assymmetric.sci	0150tc	4	0.150	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	K		
tc_assymmetric.sci	0200tc	4	0.200	17218	1538	1368	1565	2375	154	403	773	978	1-Linear	K		

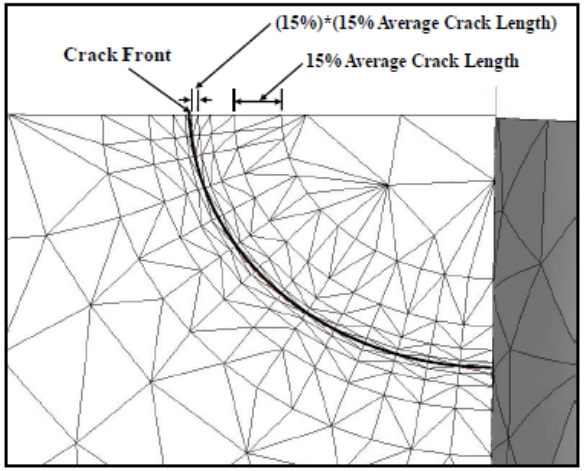
Local StressCheck FEMs Assumptions

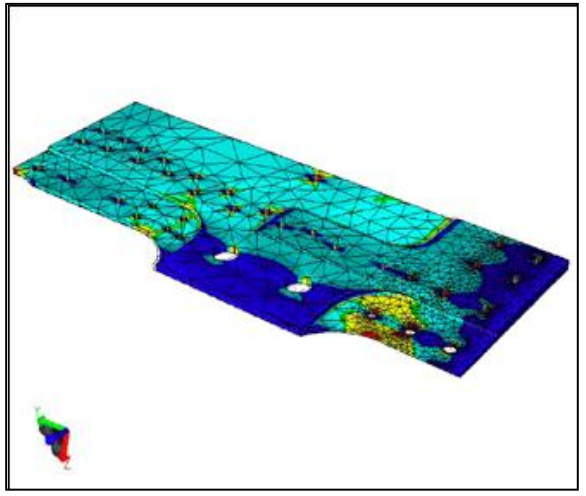
- Initial Flaw Sizes, Continuing Damage Flaw Sizes, Relative Flaw Sizes
  - Leveraged failure analyses
  - Typically leaned towards conservative assumptions
- Crack Locations
- Fixed Crack Aspect Ratio
  - Current and historic A-10 approach
- Load Redistribution Interpolation from Global FEMs

Local StressCheck FEMs

Mesh Development

- Global Automeshing
- Local Refinement Around Loaded Fastener Holes or Other Geometry Details if Necessary
- Crack Face Mesh Feature used to Simulate Crack
- Boundary Layer Refinement at Crack Front to Isolate Singularity. Ground Rules for Refinement are:
  - Crack front should have three to four layers of elements of progressive smaller size
  - The largest boundary layer elements, which are the farthest boundary layer elements from the crack front, should be approximately 15% of the average crack size
  - The first layer of elements next to the crack front should be 15% of the largest boundary elements.





Local StressCheck FEMs

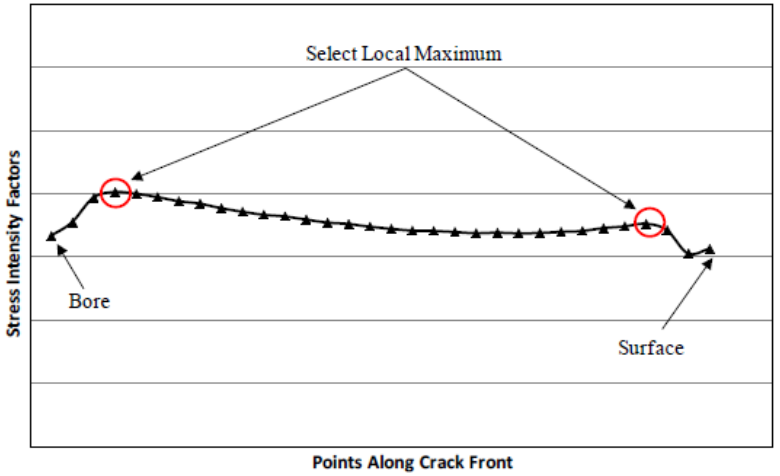
Loads & Boundary Conditions

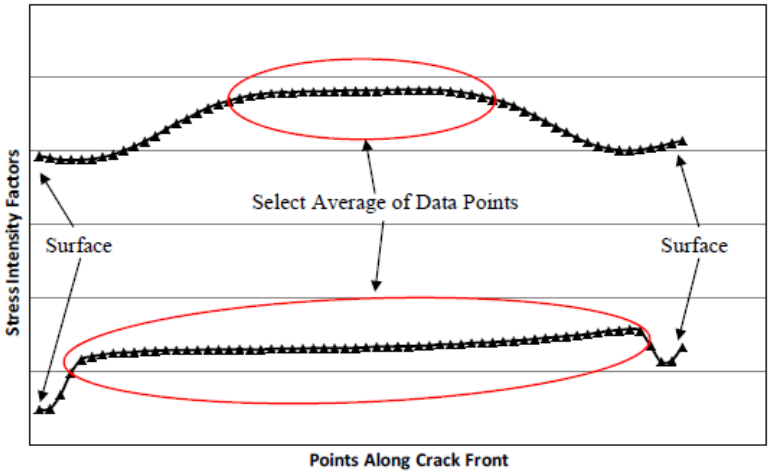
- Loads from Global FEM applied to single component StressCheck model
- Loads adjusted with changes in crack length to capture load redistribution
  - Assumptions made for interpolation between Global FEM results
- Constraints targeted to represent actual component conditions
  - Similar constraints as Global FEM

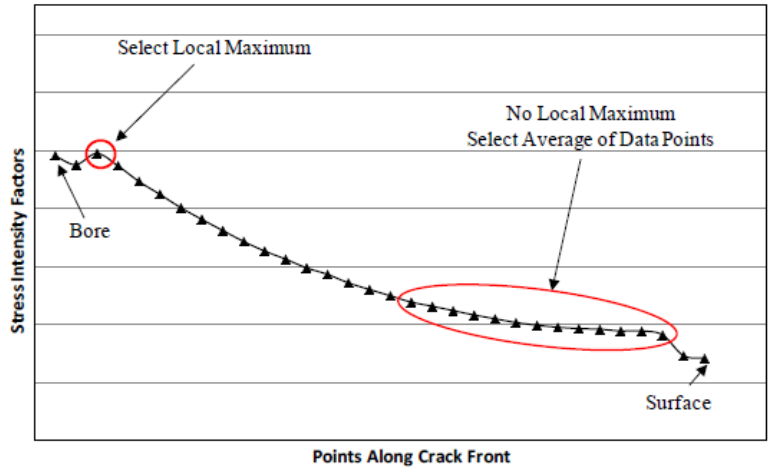
Local StressCheck FEMs

Stress Intensity Selection – Bore & Surface

- Stress intensity selection
  - Did not follow Newman-Raju 5 and 80 degrees
  - Selected local maximum (typically)
  - Focused on reasonable but conservative selections

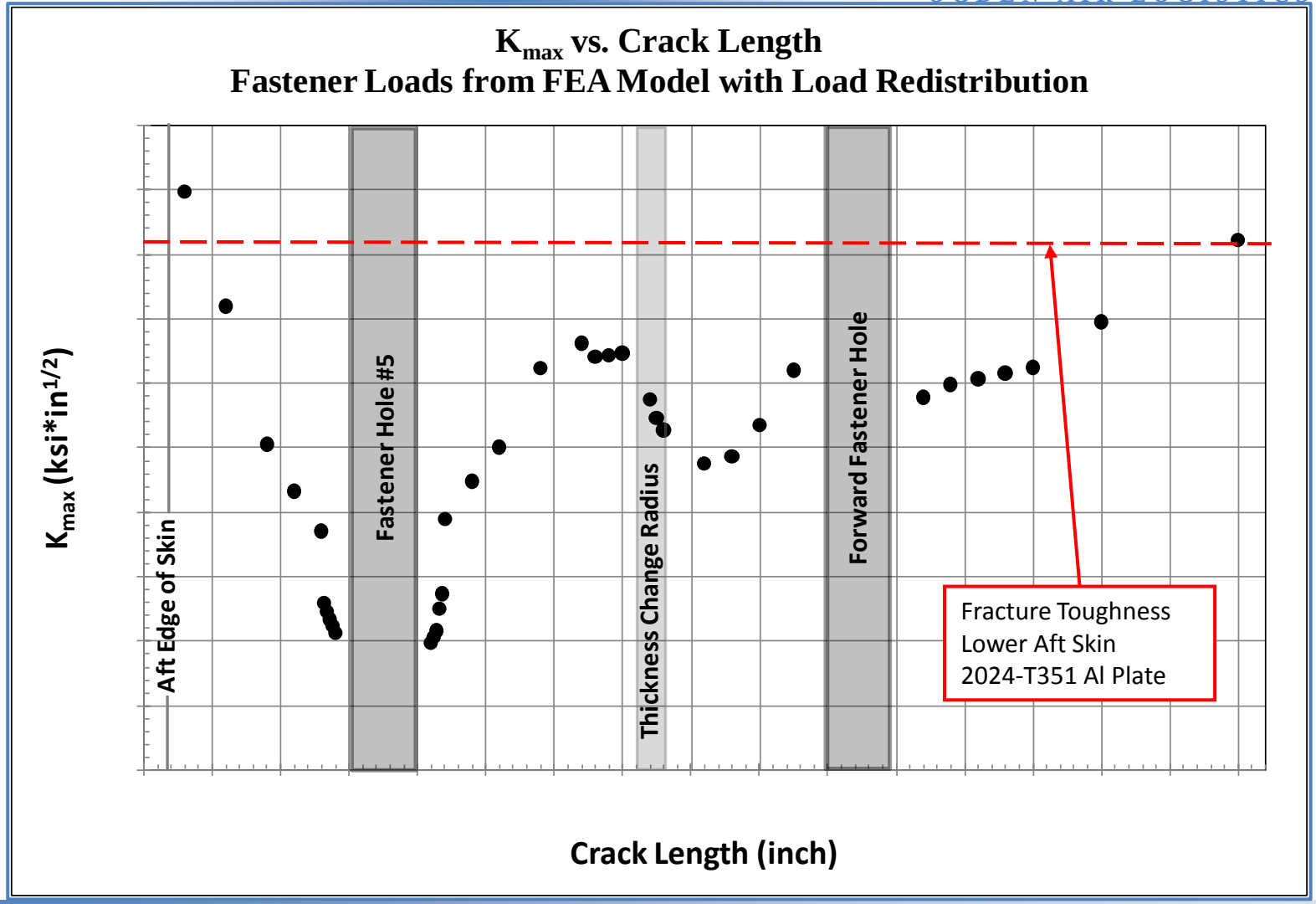




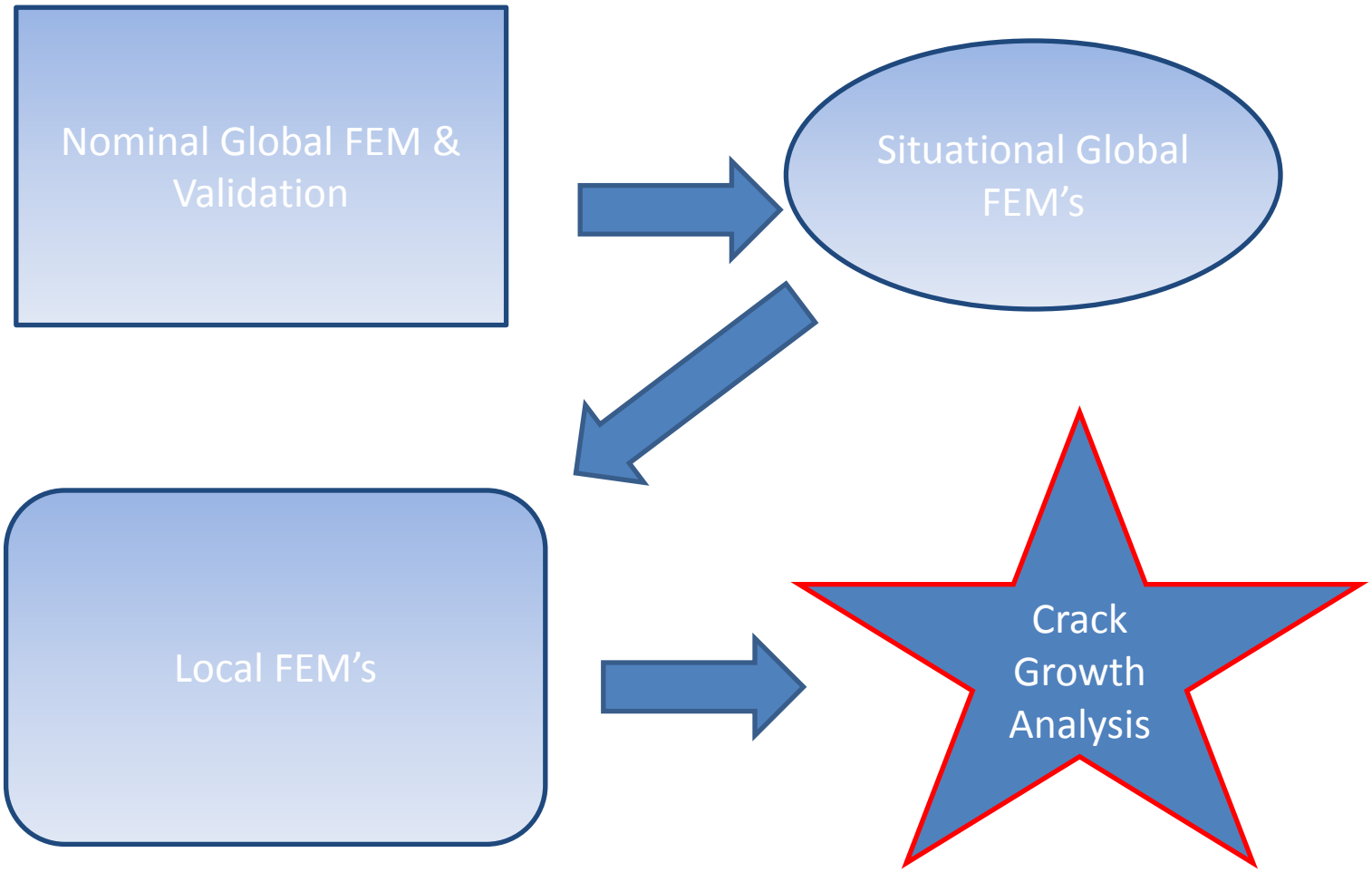


Local StressCheck FEMs

Stress Intensity Results



Global & Local FEM Flow Chart



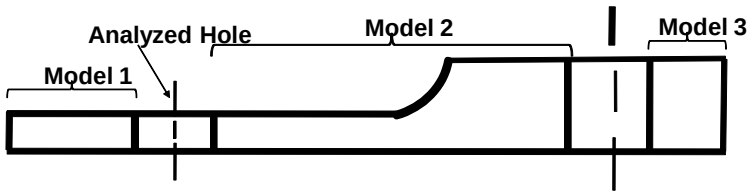
Damage Tolerance Analysis (DTA)

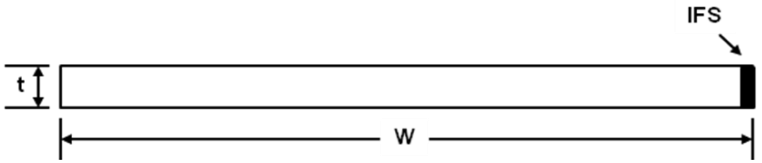
- Approach
  - Calculate beta as a function of crack length
  - Input beta & far field stress into AFGROW
  - Complex geometry created difficulty determining far field stress
    - Same far field stress used to calculate betas was used in AFGROW model
      - Determined from uncracked StressCheck FEM
      - Must be fairly accurate – retardation SOLR a function of far field stress

$$K = \sigma\sqrt{\pi a}\,\beta \qquad \beta = \frac{K}{\sigma\sqrt{\pi a}}$$

Damage Tolerance Analysis (DTA)

- Approach, cont.
  - AFGROW Single through edge crack (user defined beta)
    - Remember A/C = constant
    - Stress intensities capture effect from corner crack
  - Crack growth predictions for ligament failure, continuing damage, continuing – continuing damage, etc. is decoupled
    - Have not developed AFGROW to StressCheck plug-in capability **YET**.

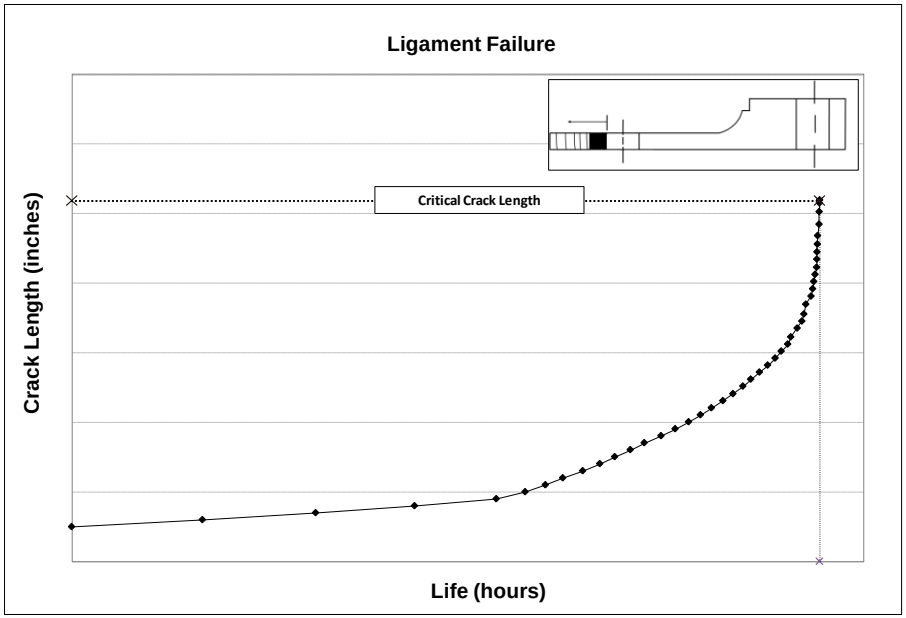


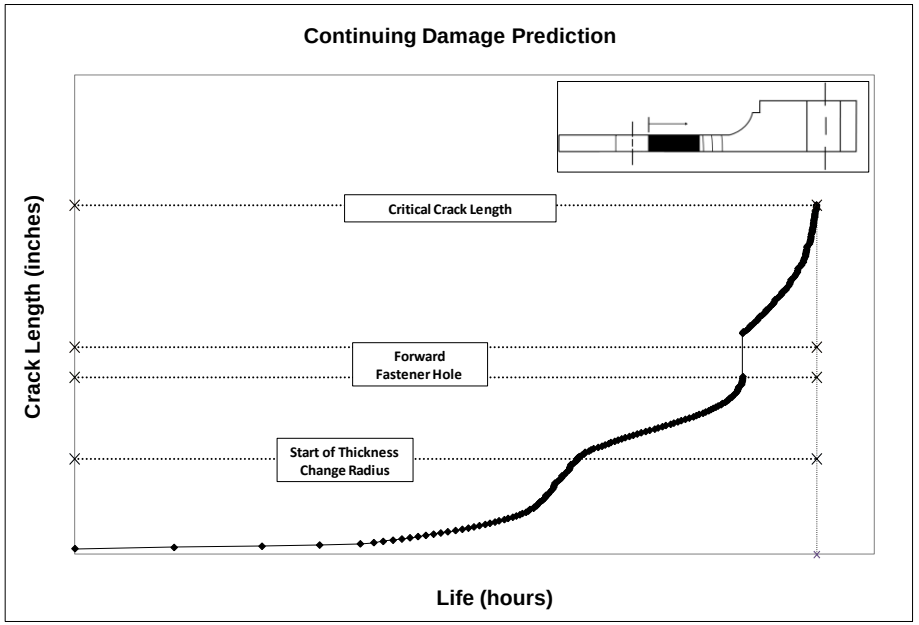


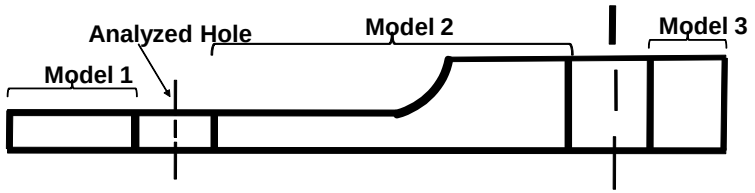
AFGROW Single Through Edge Crack
User Defined Beta Model

Damage Tolerance Analysis (DTA)

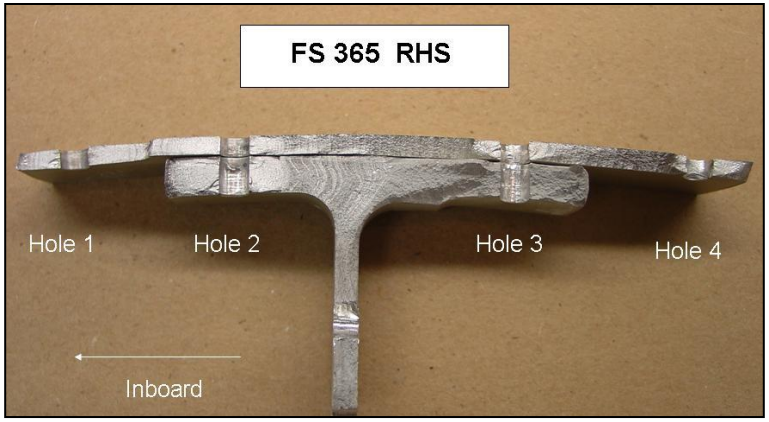
- Approach, cont.

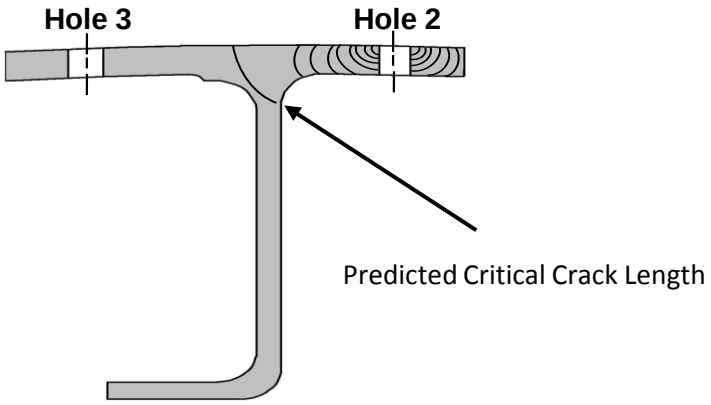


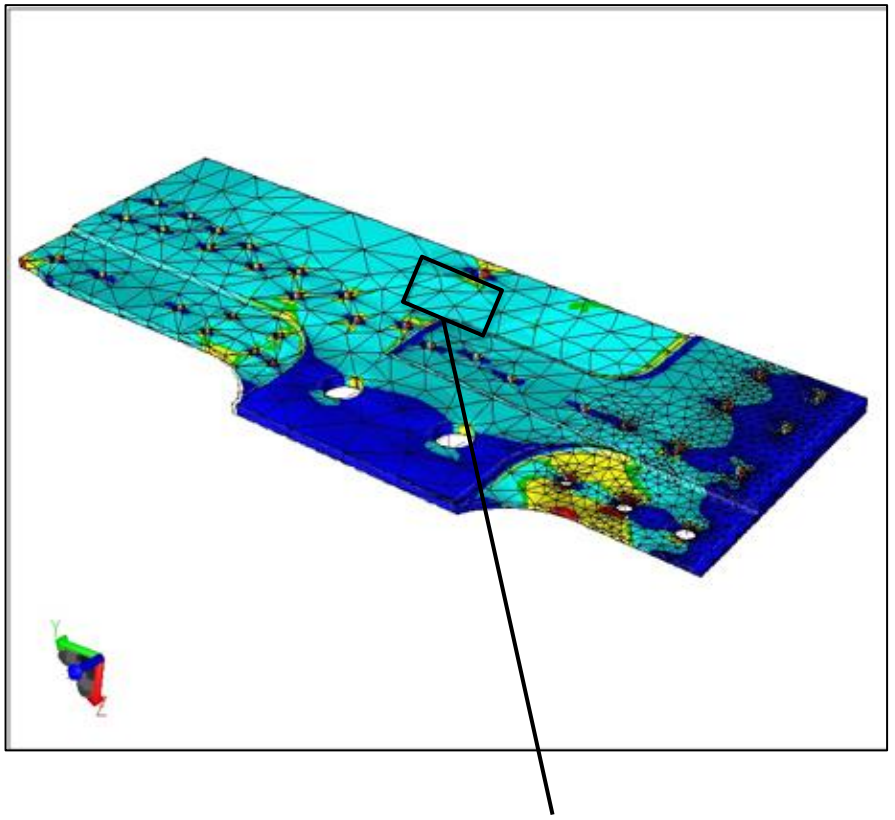




Results Validation





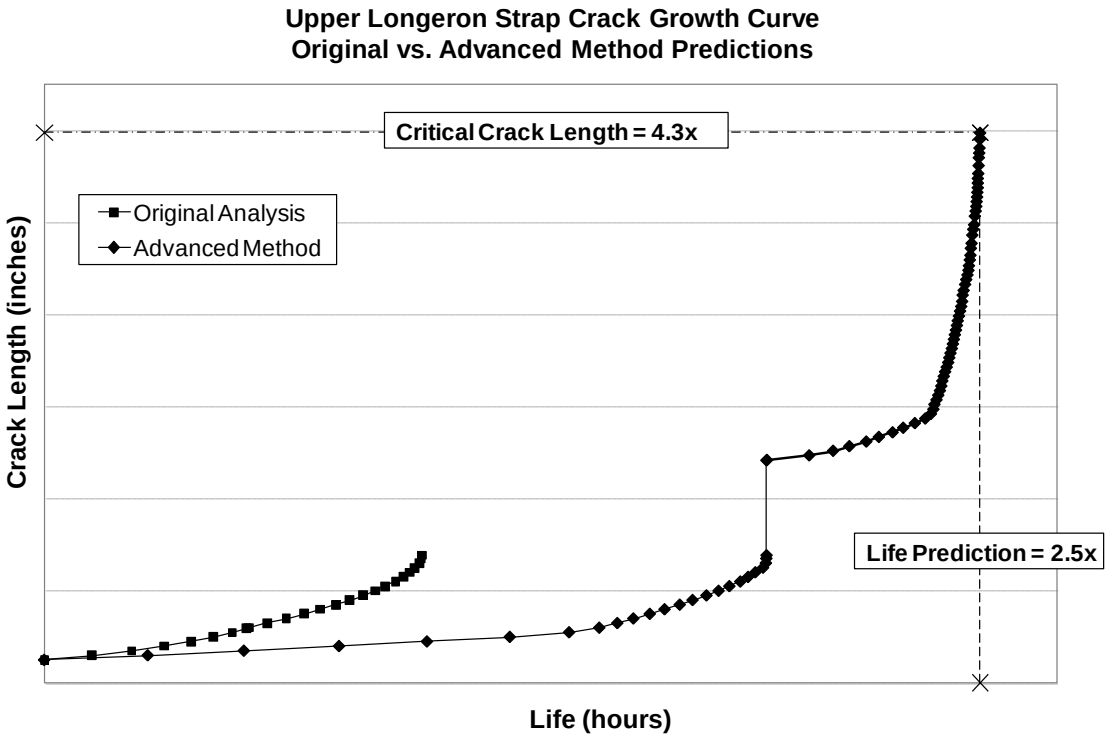


Validation of FEM Results

Strain Gage Location from Fatigue Test - within 3.3%

Why This Technique is Important

- More Realistic:
  - Critical crack length
  - Total damage tolerant life
  - Inspection requirements



How Does This Method Benefit the A-10 Fleet?

- Increased Organic Capability
- Increased Fidelity in Analysis
- More Reasonable Timeline to Address Issues
  - Typical analysis may say “Ground the Fleet”
  - Advanced analysis provides increased inspection time to allow for inspection and repair
- Decreased maintenance cost
- Increased aircraft availability

Way Forward

- Entire Aircraft FEM
  - Input detailed FEM into full aircraft model
  - Investigation of Global FEM methods to increase solving efficiency
- Plug-in Capability between SC and AFGROW
  - Next step to model multiple crack evolution all at same time
  - Standardized SC COMs and users guides

Questions?