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Nonlinear and Progressive Failure Aspects of Transport Composite Fuselage Damage Tolerance

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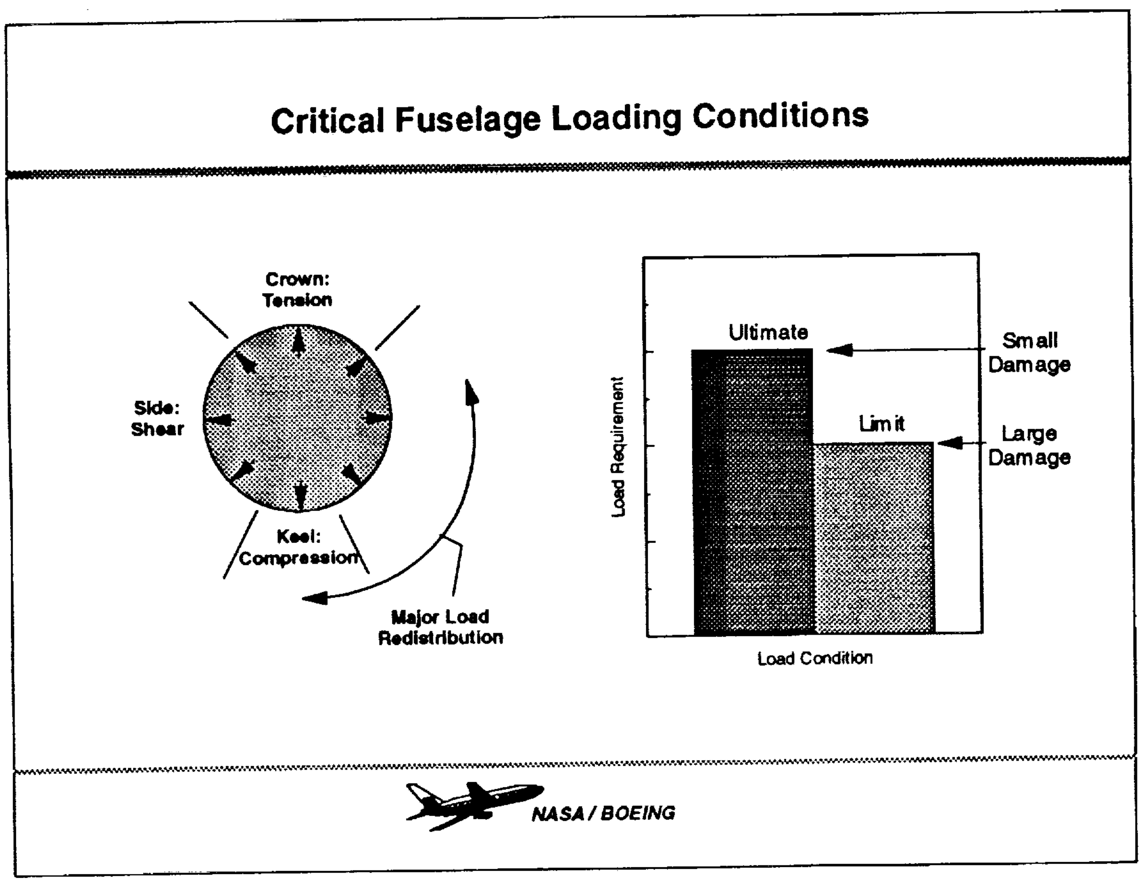
INTRODUCTION

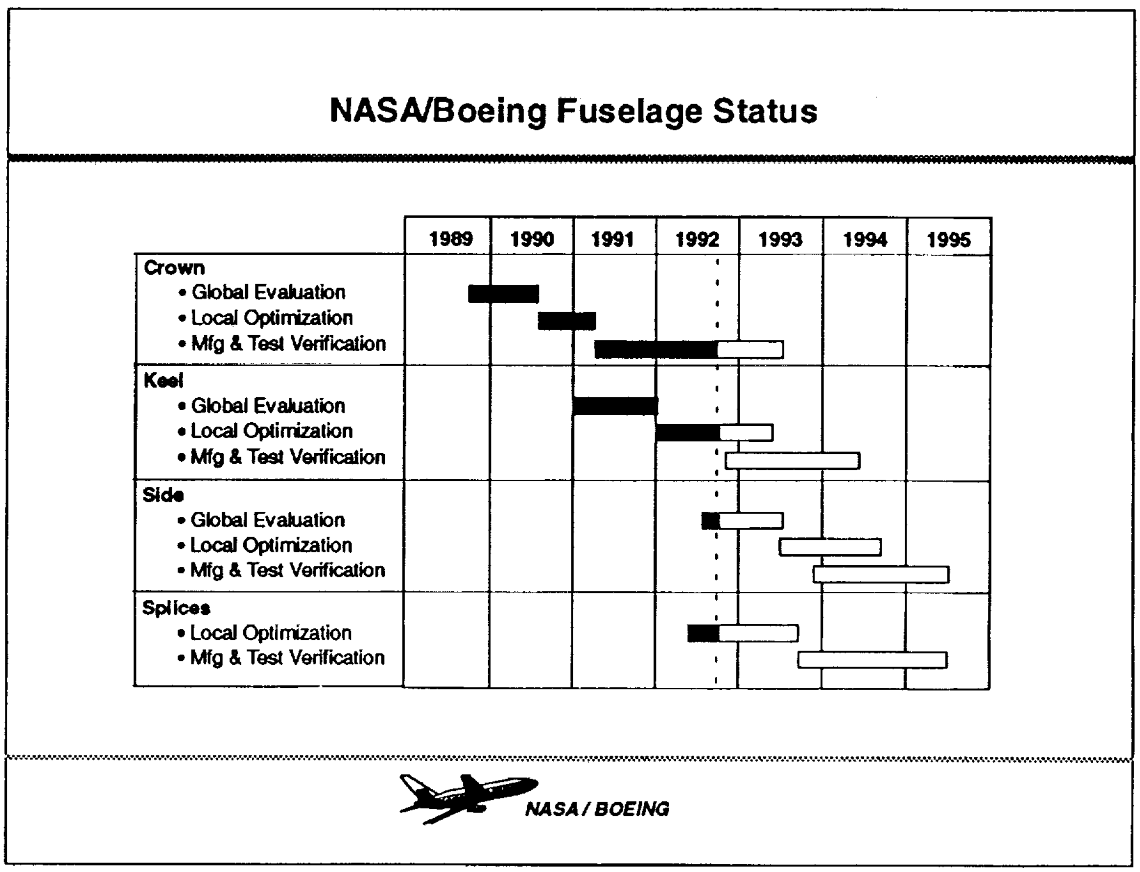
The purpose of this paper is to provide an end-user's perspective on the state of the art in life prediction and failure analysis by focusing on subsonic transport fuselage issues being addressed in the NASA/Boeing Advanced Technology Composite Aircraft Structure (ATCAS) contract and a related task-order contract. First, some discrepancies between the ATCAS tension-fracture test database and classical prediction methods will be discussed, followed by an overview of material modeling work aimed at explaining some of these discrepancies. Finally, analysis efforts associated with a pressure-box test fixture will be addressed, as an illustration of modeling complexities required to model and interpret tests.

Fuselage loading is complex, with combined loads in all regions. ATCAS has divided the cylinder into four quadrants based on primary loading considerations. The internal pressure is reacted primarily as hoop tension, and is effective in all quadrants. Critical axial loads are primarily tension in the crown and compression in the keel, with shear being dominant in the side. The upper and lower portions of the side panel have significant regions of combined tension-shear and compression-shear, respectively. The lower side has the additional issue of major load redistribution around cargo door and wheel-well cutouts.

Load levels are necessarily coupled with damage states. Ultimate load levels must be sustained with undetectable damage, the upper limit of which is often "barely-visible damage." Limit loads must be sustained with large damage levels, often represented in tests with element and/or skin saw-cuts. The prediction of strength with damage caused by realistic threats is complicated by the limitations of current non-destructive inspection methods to accurately quantify existing damage states.



The ATCAS schedule indicates the current status. Crown activities are nearing completion, with only component tests remaining. The keel and splice activities are in the technology development stage, and the side efforts are addressing design trade studies. Further discussions will focus on the crown region since it is the farthest along. The problems raised are representative of what has been found, and have some general applicability. Additional issues are likely to be uncovered as the keel, side, and splice regions are addressed in more detail.



CROWN PANEL TESTS VERSUS EXISTING THEORIES

The topic of the following discussions will be limited to tension damage tolerance, which is a critical design driver in the crown region. Several competing failure modes contribute to this issue, including skin fracture, stiffener strength, and skin/stiffener attachment. Each is affected by several variables. In addition, behavioral characteristics of composite materials that must be contained in predictions include damage growth simulation, trade-offs between strength and toughness for laminate/material variations, and load redistribution. The competing failure modes interact through load redistribution. For example, as stable damage growth occurs in the skin, additional load is projected towards the stiffener, requiring additional load-transfer capability in the skin/stiffener attachment and additional load-carrying capability in the stiffener. Similarly, limited amounts of debonding or fastener yielding along a skin/stiffener interface as major damage approaches provide a structural benefit, shielding the stiffening element from the sharp stress concentration in the skin. More severe debonding or fastener yielding, however, is detrimental, removing the stiffening element from the structural load paths. Understanding and having predictive capabilities for these complex interactions are essential to developing balanced structural designs.

Crown Panel Damage Tolerance Example

Competing failure modes

- Skin fracture
  - Layup
  - Material
  - Manufacturing process
  - Load rate
  - Environment
- Stiffener strength
  - Layup
  - Material
  - Load rate
  - Environment
  - Damage state
- Skin/stiffener attachment
  - Nonlinear shear stiffness
  - Load sharing
  - Fastener flexibility
  - Bondline strength

Unique characteristics of composite materials

- Damage growth simulation
- Strength versus toughness
- Load redistribution

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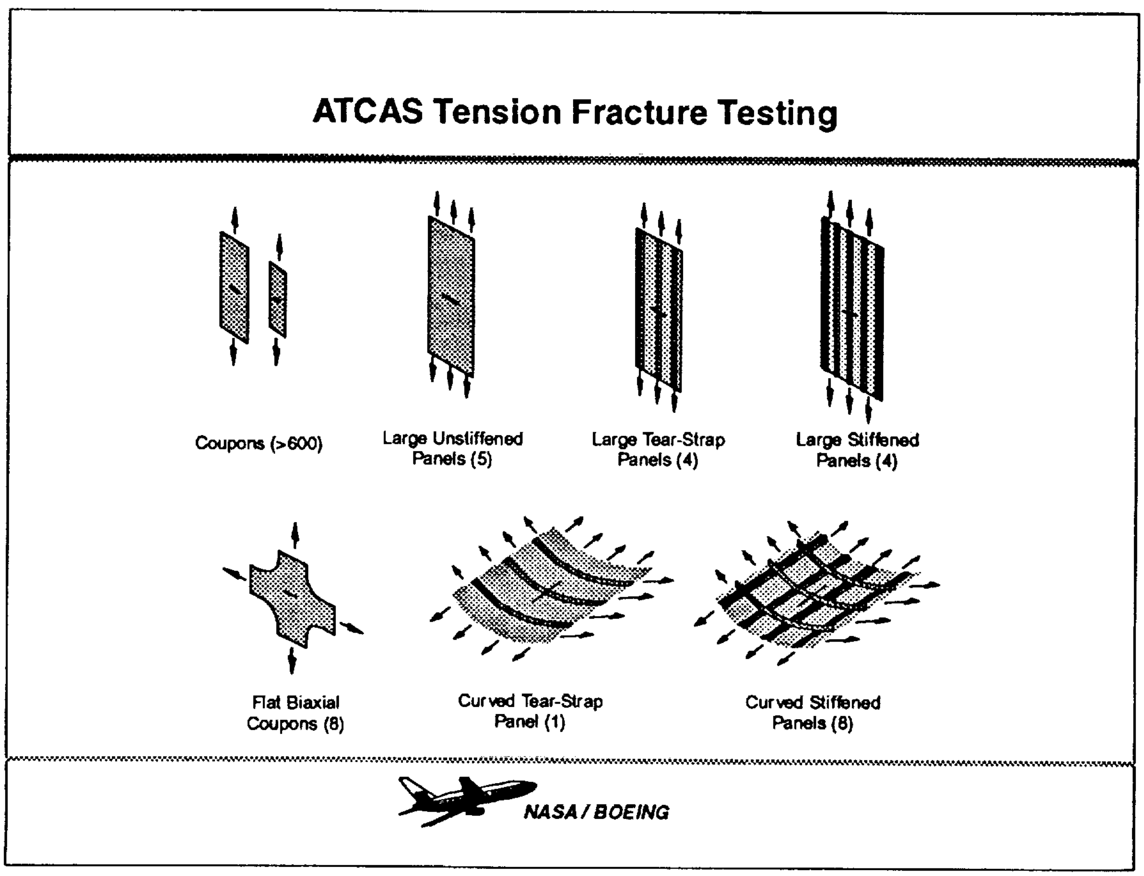
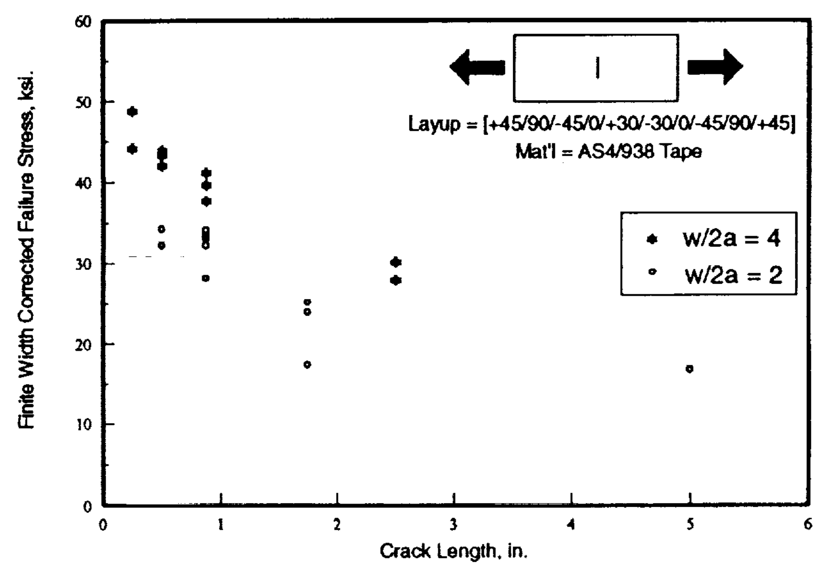
ATCAS has obtained a large tension-fracture database, ranging from small coupons to 5' x 6' fully configured panels. The wide range of variables included in the testing have proven to be extremely valuable in understanding analytical limitations. The database is being thoroughly documented, and is available for verification of predictive techniques. The following discussions focus on the relationships between simpler specimen geometries and analyses. Any difficulties encountered at this simple level will be magnified as more complex structure is addressed.



Classical methods have been found to underpredict the effects of specimen geometry. The figure contains two sets of data, each with a different specimen-width-to-notch-length ratio (w/2a). Both data sets have been corrected for finite width using classical finite width correction factors (FWCF), and should fall on a single curve if the FWCFs accurately predict the geometry effects. The two distinct curves indicate the inaccuracies of classical FWCFs. Similar results were observed for other laminates, materials, and less severe specimen geometries (i.e., between w/2a = 4 and w/2a = 8). The inaccuracies are caused by larger-than-expected projections of stresses towards the specimen edge. This projection is likely caused by a combination of (a) prefailure damage progression, (b) transverse buckling of the notch, (c) repeatable material inhomogeneities, and (d) point-rotation degrees-of-freedom in the material.

Finite Width Effects



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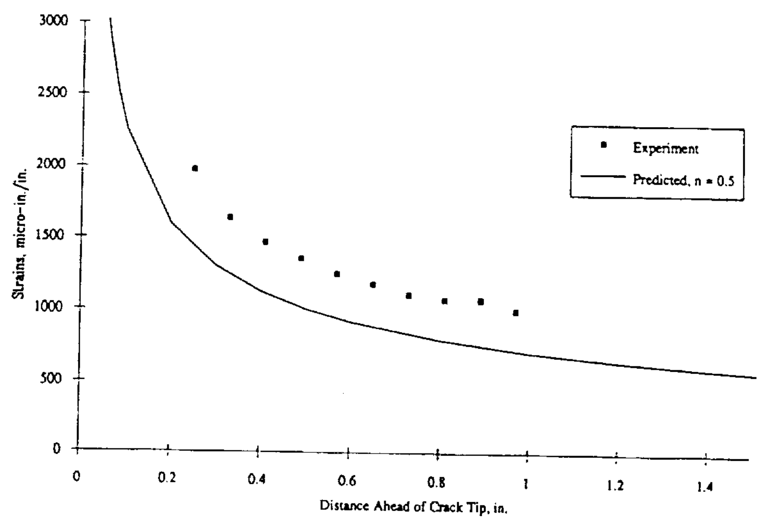
In large notched specimens, a projection of strains towards the specimen edge was observed prior to any damage growth from the notch tip. For this particular laminate/material combination, classical square-root-singularity methods underpredict the actual strains by approximately 25%. Similar trends, with similar or smaller differences between predicted and measure strains, were observed for a variety of other laminate/material combinations.

Pre-damage Notch-Tip Strains



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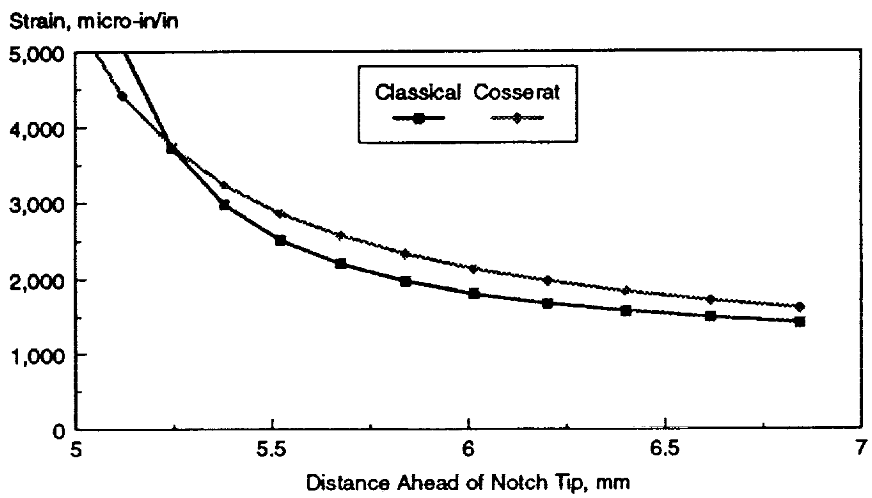
Dr. Roderick Lakes, on sabbatical to Boeing from the University of Iowa, illustrated similar strain projection using Cosserat material models. These models allow for point-rotation degrees of freedom in the continuum. It should be emphasized that it is important to accurately predict strains prior to damage growth, since load distribution is critical in predicting both structural response and local failures. Models that do not predict initial distributions will not accurately follow the redistribution as damage progresses, and will therefore be unable to predict each of the competing failure mechanisms. Note that the distributions, and therefore effects of specimen geometry, for Cosserat theory differ as a function of the material length parameters.

Cosserat Crack-Tip Strains



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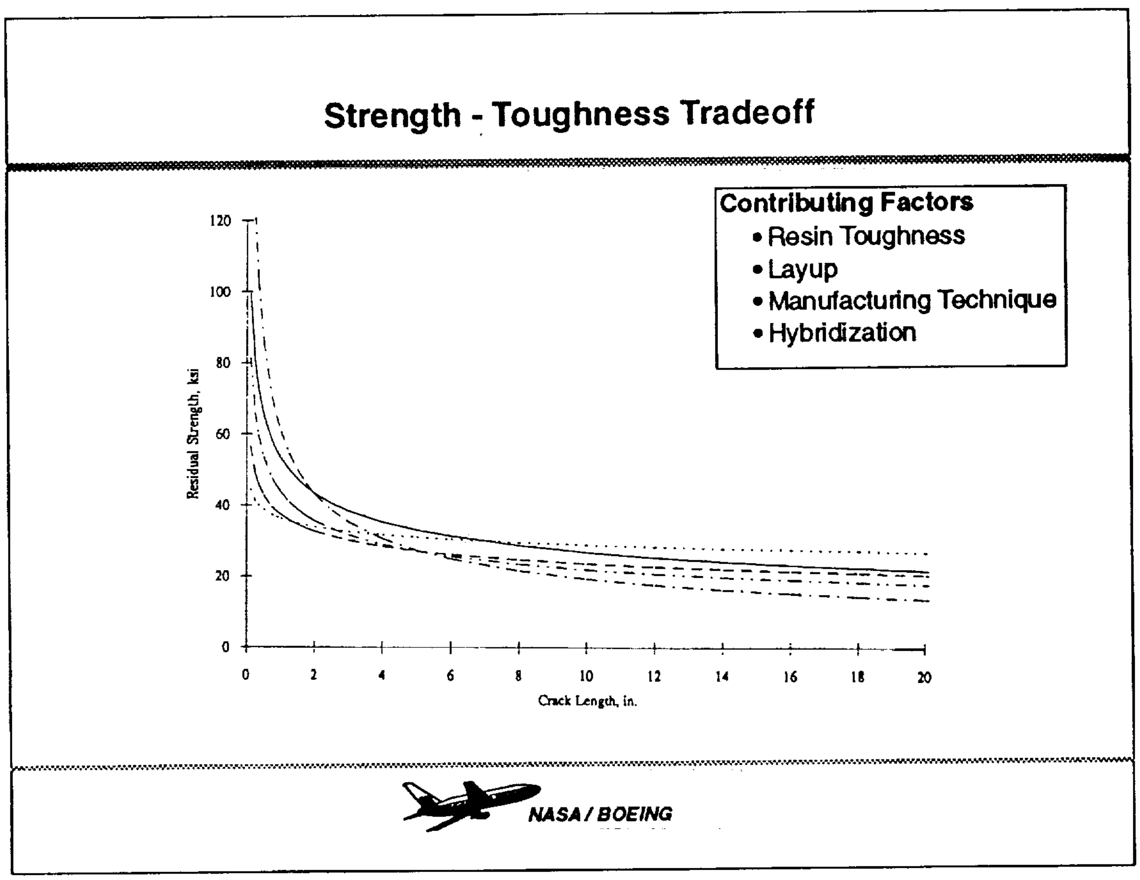
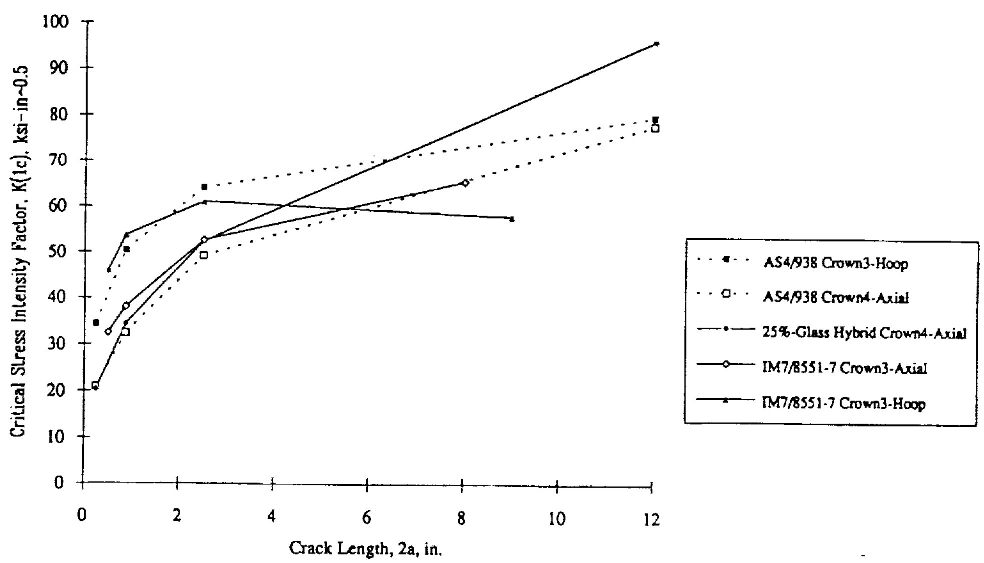
Significantly different residual strength curves were observed in the ATCAS tension fracture testing. Variables in this behavior include resin toughness, layup, manufacturing technique, and hybridization. The differences imply that a trade-off between strength (small-notch strengths) and toughness (large-notch strength) exists, as is the case with aluminum alloys (e.g., 7075 vs. 2024). Tough resins, hard layups, and small scales of repeatable material inhomogeneities result in higher strengths but lower toughnesses. Conversely, brittle resins, soft layups and large scales of repeatable material inhomogeneities result in lower strengths and higher toughnesses. The higher-strength, lower-toughness laminate/material combinations tend to follow classical predictions more closely. The lower-strength, higher-toughness materials respond as would be predicted for a reduced-singularity stress field.



In fact, the higher-strength, lower-toughness combinations converge to their classical mode 1 stress intensity factor (K_{1c}) at smaller notch sizes than do the lower-strength, higher-toughness combinations. It should be noted that the toughest laminate/material combinations, which are most attractive for skin applications, do not converge until well into the crack-size range of interest. Classical fracture mechanics properly predicts failure of a particular laminate/material combination for all notch sizes within the converged-K_{1c} range. For notch sizes below the converged-K_{1c} range, prediction of specimen failure becomes analogous to an elastic collapse problem.

Convergence of Stress Intensity Factor



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STRAIN SOFTENING MODEL DEVELOPMENT

After careful review of many previous efforts to analytically simulate and predict the fracture characteristics of laminated composite materials, a sophisticated nonlinear finite element implementation of the cohesive stress crack theory has been undertaken. Relative to metallic structure, the nonlinear softening behavior that occurs in the vicinity of a crack in multidirectional composite laminates involves a much larger area. Experimental observation suggests that the damage zone at a crack tip in composite laminates is large enough to be represented by several finite elements in a model that can be economically and quickly processed.

Extensive experimental study strongly suggests that a comparatively large damage zone develops around cracks in laminates and that a number of physical phenomena contribute to a strain softening effect

- Fiber breaks
- Matrix cracking
- "Scissoring" of angle plies
- Crack bridging, fiber bundle pull-out

By introducing a local, non-monotonic load capability (elastic, yield, unload) to a finite element model, a damage zone of finite size is represented and stable crack growth can be simulated

The resulting problem is extremely nonlinear, both locally and globally, and has been solved using the ABAQUS analysis system

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A flat, center crack tension coupon is modeled using two planes of symmetry. Initial studies assumed self-similar crack growth, allowing the loading, yielding, and unloading along the crack line in the model to be precisely prescribed with individual spring elements. The load-displacement relationships for these springs are derived from the measured stiffness and failure strengths of the laminate/material combinations being studied.

Problem Formulation

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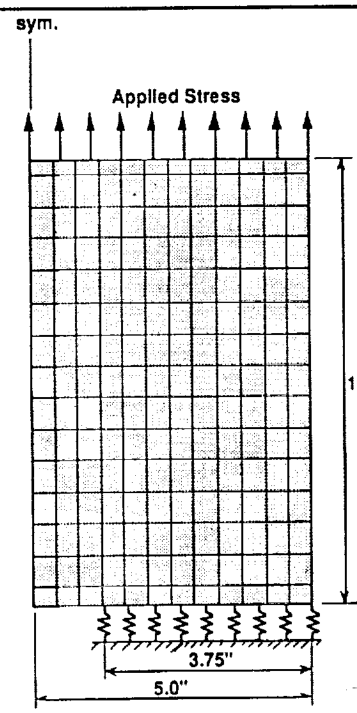

A detailed analytical study using design-of-experiments principles was conducted to evaluate the sensitivity of the specimen response to each of the parameters which define the strain softening law. The most dominant parameter affecting residual strength for a given notch size was found to be the maximum stress for elastic laminate behavior, σ_{max}. Other factors tended to control the shape of the residual strength curve (i.e., change in residual strength as a function of notch length).

Strain Softening Law

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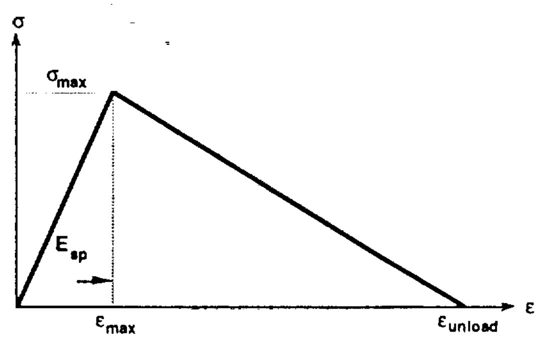

F	=	$\sigma \cdot A$
A	=	element thickness · distance between springs
σ	=	stress within spring
E_{sp}	=	spring modulus
ε_{max}	=	failure strain
ε_{unload}	=	strain at total unload

These models were exercised to determine if the proper degrees-of-freedom exist to predict the strength-toughness trade-off observed in the ATCAS test data. As shown in the figure, a softening law with a relatively short but steep unloading segment predicts a steeper residual strength curve than a law with a longer, less-steep unloading segment. Since classical materials instantaneously unload at a single strain, the steeper unloading curve is more representative of classical response, and does, in fact, result in a residual strength curve closer to that predicted by classical fracture mechanics. In addition, steep unloading curves also tend to drive a classical response in the finite element models whereby a more dense mesh is needed to facilitate failure prediction. These findings illustrate that the proper degrees-of-freedom required to predict the observed response are present, and there appears to be a physical basis for the observed predictions.

Influence of Strain Softening Laws

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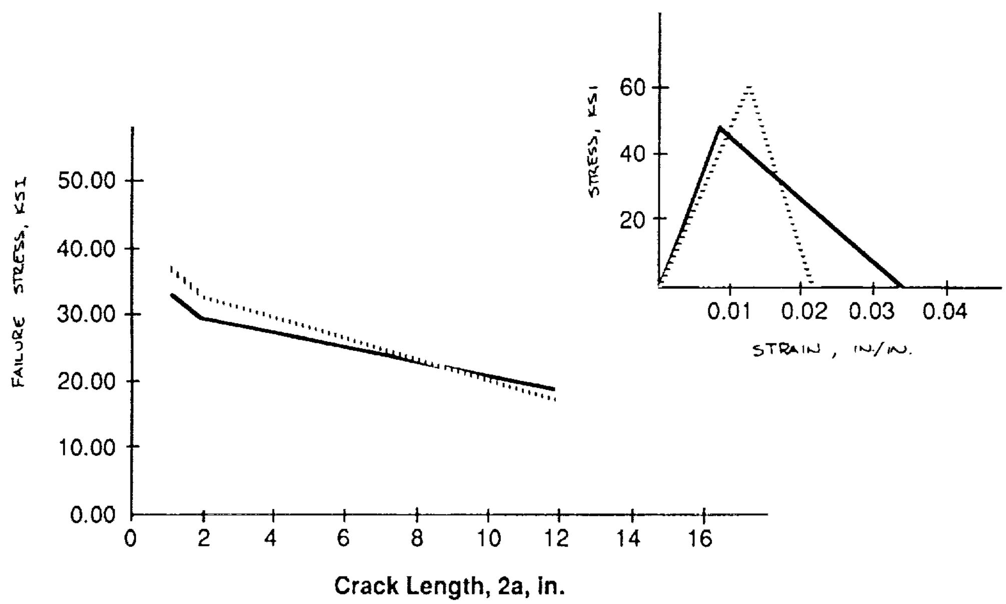

The specific laminate/material combinations tested in ATCAS were analyzed using this strain-softening approach to evaluate the accuracy of predicting their residual strength curves. As this figure illustrates, linear elastic fracture mechanics (LEFM), calibrated to 2.5" crack test results, grossly overpredicts fracture strength for smaller cracks, and underpredicts test data by 40% at larger crack lengths. Applying the damage zone model (also calibrated at the 2.5" crack test results) results in significantly improved predictions of actual response.

With large-damage conditions controlling much of the crown design, any conservatism in the anticipated strengths at these notch sizes translates directly into additional design cost and weight. Minimizing the conservatism can be accomplished either by testing the large notch sizes (a costly proposition) or predicting the large-notch strengths by analytically extending the small-notch strengths. The strain-softening models clearly provide superior extrapolation capability, and also predict the load redistribution resulting from damage growth that is required for accurate structural analysis.

Comparison of AS4/938, Crown4 Laminate – Axial Data

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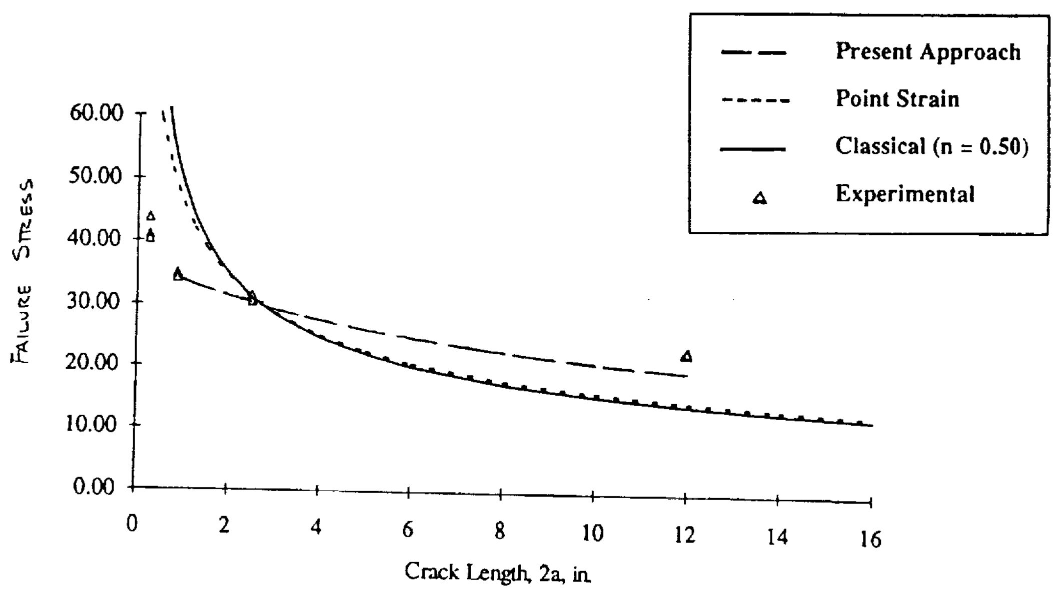

Another significant predictive ability demonstrated by the damage zone model is the sensitivity of center crack test specimens to the width of a coupon relative to the crack size (w/2a). A single strain-softening law was obtained by calibrating at a single specimen geometry, and used for all other geometries. As can be clearly seen, the strain-softening law predicts differing trends between the two data sets. This initial attempt at predicting the experimentally-observed differences resulted in surprisingly good correlation with the data. Further understanding of the effects of the law parameters on the response would likely lead to improved correlation.

Comparison of Finite Width Correlated Strength

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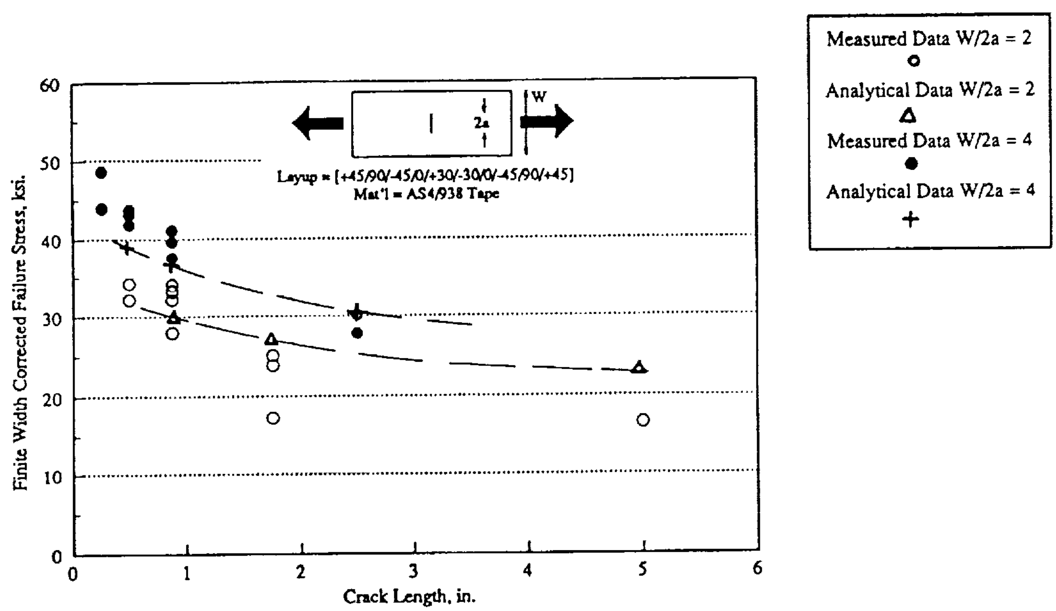
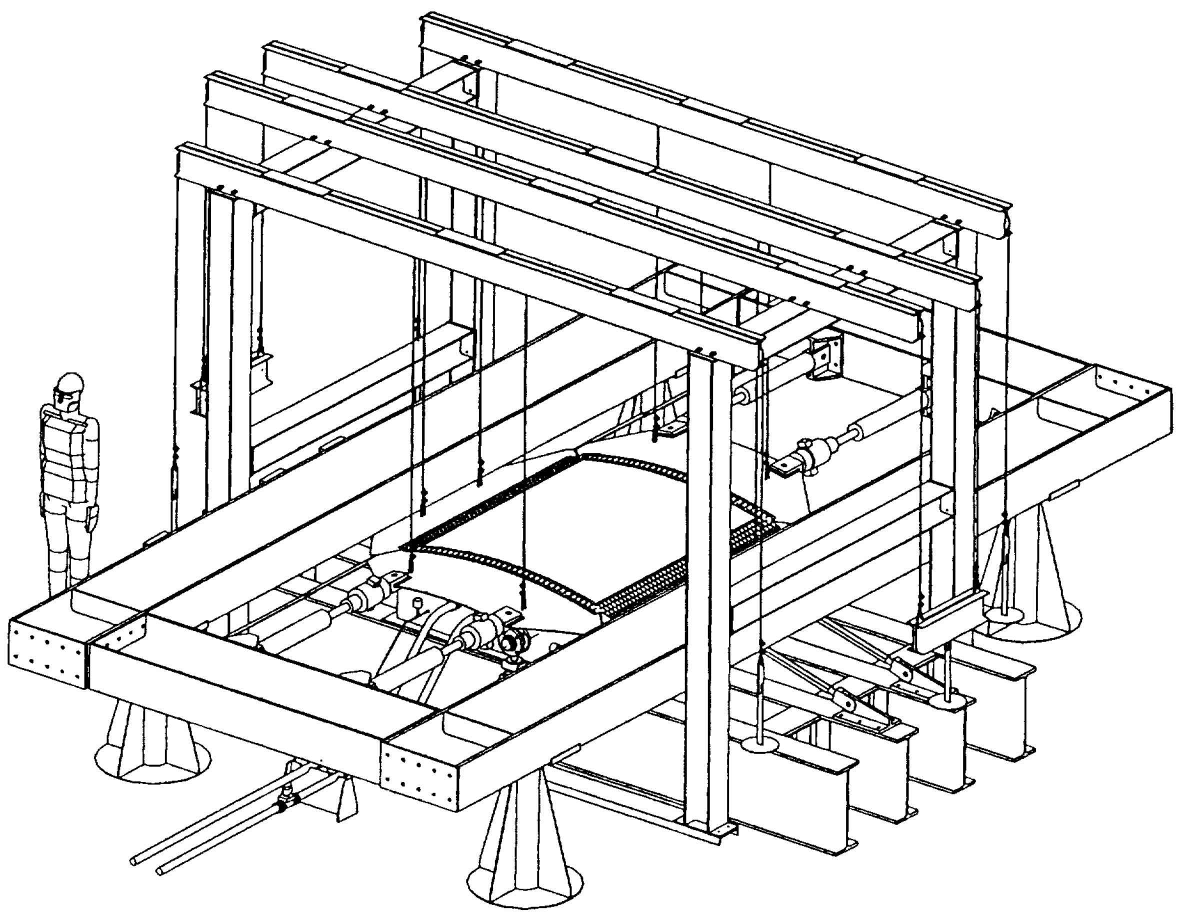


ANALYSIS REQUIREMENTS FOR COMBINED LOAD TEST FIXTURE

The purpose of the pressure-box test apparatus is to simulate the structural response of a portion of a 122-inch radius aircraft fuselage. The tests specimen is a 72 in. x 63 in. graphite/epoxy skin panel with the curvature along the 63 in. edge. The test fixture permits the inclusion of longitudinal stiffeners and circumferential frames. The heart of the test fixture is the pressure box, which permits the simulation of a pressurized fuselage. Pressure loads act on the skin panel and are reacted in the hoop direction by large plates attached to the skin panel and by truss elements attached to the frames. Axial loads arising from internal pressure and/or fuselage bending are introduced by hydraulic cylinders attached to axial loading plates. The test specimen, the loading plates, and the various reacting trusses and actuators are free to float on the pressurized air.



Finite element analysis of the pressure-box test apparatus has played an ongoing role in its design and development. Initial analyses were focused on sizing and locating the test fixturing to most accurately represent the remainder of the fuselage. These items included the loading plates, the pairs of grips which transmit load between the test specimen and the loading plates, the actuators, and the load reaction members. As the design has matured, more detail has been incorporated in the finite element model in efforts to finalize the panel doubler configurations to minimize interactions between the test fixturing and the panel response. The predictions may also serve to identify and quantify any discrepancies which might be unavoidable.

Objectives

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Geometrically nonlinear (large deflection) finite element simulations of the pressure box test fixture have been performed using ABAQUS. The objectives of these analyses are

1. Identify and understand the interactions between the test fixture and the stringer and frame stiffened test panels. This task is critical in relating pressure box test data to full fuselage analyses, and to larger scale tests. (Scaling)

2. Support the detailed design of the pressure box test apparatus

3. Generate pretest predictions and recommend sensor locations

4. Evaluate the effects of damage to pressure box test panels subjected to biaxial loading. Compare to test results and complete fuselage assessments. (Damage scaling)

The current finite element model represents one quarter of the test specimen, as shown in the figure. It consists of 2260 node points, 238 beam elements, and 1911 shell elements. This model requires approximately 90 seconds of CPU time to run a static, large deflection analysis on the Boeing CRAY YMP. When damage is represented in the model, local deflections are much larger, and run times increase to approximately 175 seconds.

Detailed Finite Element Model

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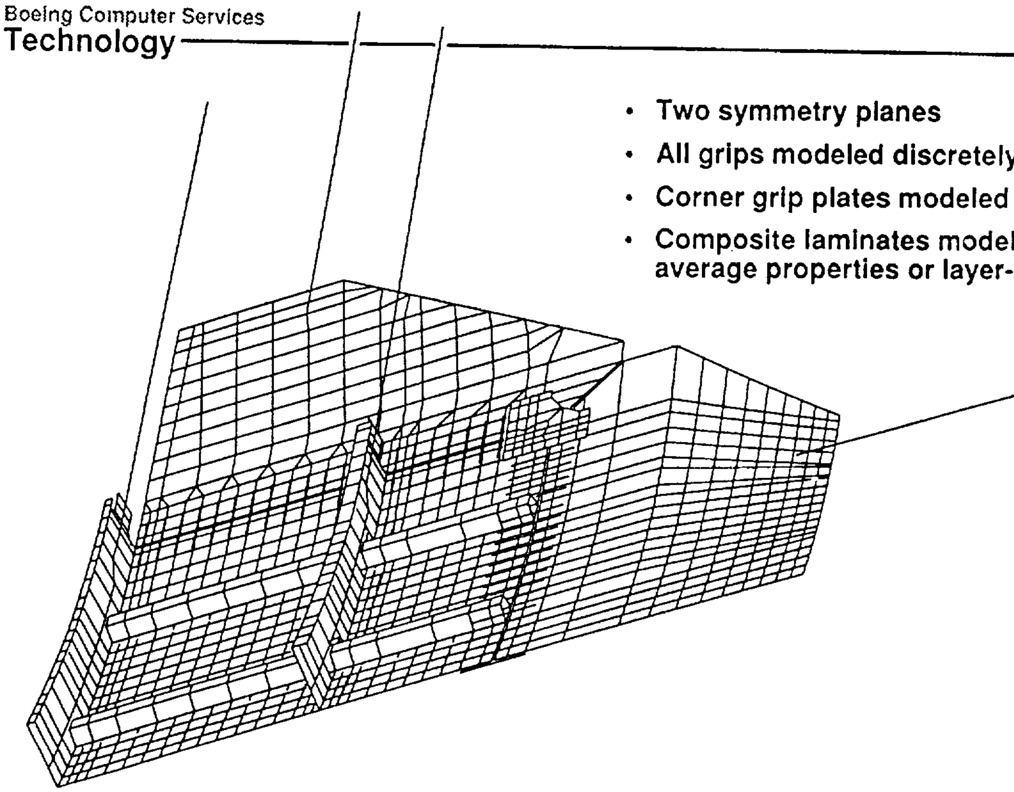
- Two symmetry planes
- All grips modeled discretely
- Corner grip plates modeled in detail
- Composite laminates modeled with average properties or layer-by-layer



A great deal of effort has been devoted to a more detailed representation of the pairs of grips and the corner plates which transfer loads between the test specimen and the loading plates. To permit the grips and plates to be modeled at their proper radii relative to the plates and test specimen, and to duplicate the slotted attachments of the hoop grips, the beams representing the grips must be joined to the model via multiple constraint laws and special equations. This has permitted a more accurate assessment of the "as built" test hardware.

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The finite element models have evolved with and supported the detailed design of the pressure box test apparatus. Simulations of various load cases have been used to assess and verify:

- Grip design and resulting load distributions
- Location of axial load application
- The effectiveness of "slotted" hoop grips and corner grip plate design
- Grip fastener loads – led to the step-tapered, "scalloped" doubler design for introducing axial loads
- The impact of grip and doubler design on the response of panels with central damage

{Approximately 8 man-month support effort}

SUMMARY

NASA/Boeing programs are generating a large structural database and supporting analysis methods for composite fuselage damage tolerance. Crown test results indicate that advanced analysis methods are needed to simulate composite failure. Strain-softening models have been successfully implemented in an existing nonlinear finite element code (ABAQUS), and have predicted several important experimentally-observed trends that were not properly addressed by classical fracture mechanics methods. Evaluation of the pressure-box test fixture have indicated that detailed analyses of the panels, attachments, and test fixtures are required to properly interpret test results.

RECOMMENDATIONS

At Boeing, the goal of composite materials research is the critical assessment of the cost- and weight-efficiency of using advanced composite materials in commercial aircraft primary structure while ensuring structural integrity. To guarantee competitiveness, it is essential that the assessment and subsequent application of attractive concepts be accomplished as expeditiously as possible. This is most efficiently accomplished with a combination of test and analysis. Test data is currently essential for real-world engineering problems that include built-up structure, major load redistribution (e.g., around cutouts), combined loads, and damage tolerance. Analysis can play a role in extending element and subcomponent panel test data to structural design problems. Analytical developments over the next five to ten years should be dominated by these concerns, since the prohibitively high costs associated with a test-only certification approach are the large scale tests for multiple load, damage, and environmental conditions.

Analysis methods can also be used to (a) reduce testing requirements and hence developmental, verification and certification costs; (b) reduce response time in the resolution of field problems and identify sites for periodic inspection; (c) support concurrent engineering problems in which manufacturing desires design details that require comprehensive analysis to ensure performance is not compromised; and (d) support composite material development through a basic understanding of failure mechanisms and their sensitivity to design variations. These applications should be longer-term goals for analysis method development.

To develop methods for extending element and subcomponent test results to fully configured structure, it is important to focus efforts to real-world problems, including layups, structural configurations, loading conditions, and damage requirements. A method to accomplish this is shown in the figure. The thrust of this approach is to identify "benchmark" problems to identify the configurations, loading, and associated technical issues. Attractive material forms would then be evaluated under loading conditions representative of their structural usage. Next, the structural role of each of the elements should be evaluated under representative loading conditions. Finally, the response of the structural assembly under flight loads and damage tolerance conditions would be assessed. With the knowledge gained in each step, the evaluations in the previous steps would be revisited, and modification made where required. Each step must include a combination of test, analysis, and documentation of the findings. Analytical capabilities must be assessed by their ability not only to predict failure, but to predict structural response throughout the loading regime.

Approach to Failure Analysis Development Supporting Composite Aircraft Structure

Identify BENCHMARK PROBLEMS and associated technical issues for analysis development.

- representative structural details for economically feasible concepts (design configuration, manufacturing methods, material forms)

Evaluate mechanical response of MATERIAL FORMS under representative loads

- modulus, damage growth, failure

Evaluate function of individual STRUCTURAL ELEMENTS under representative loads

- stiffness, strength, fracture, load redistribution

Evaluate combined response of elements in STRUCTURAL ASSEMBLY under flight loads and damage tolerance requirements

- Y + C factor analysis

Each step must include test, analysis, and documentation of findings

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From the work conducted to date at Boeing on tension fracture of fuselage crown structure, several detailed recommendations can be made. Non-destructive evaluation methods should be developed for quantifying composite material response and damage states. Lamb-wave dispersion has proved attractive in measuring damage levels near impact locations and in progressive damage zones. Methods for determining operative generalized constitutive laws (e.g., non-local, Cosserat) and their necessary material constants are also required.

Further work should be pursued on extending strain softening models for composite structural analysis. This activity includes addressing an orthotropic plate element, including membrane, shear and bending laws. Inclusion of Cosserat material behavior might be necessary. Other issues include (a) development of softening laws for compression and combined loads, (b) environmental and dynamic effects, and (c) analysis and test schemes for multidirectional material characterization.

Needs exist for development of a larger element and subcomponent test database, including shear lag, combined loads, and major load redistribution. Further analysis of subcomponent combined load tests (i.e., pressure-box tests) is required, including a range of panel design details and damage states. An analysis of the full-scale fuselage subjected to the full range of loading conditions and damage states is necessary to allow evaluation of the subscale tests.

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