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Abstract

A comprehensive analytical methodology has been developed for predicting the onset of widespread fatigue damage in fuselage structure. The determination of the number of flights and operational hours of aircraft service life that are related to the onset of widespread fatigue damage includes analyses for crack initiation, fatigue crack growth, and residual strength. Therefore, the computational capability required to predict analytically the onset of widespread fatigue damage must be able to represent a wide range of crack sizes from the material (microscale) level to the global structural-scale level. The results of carefully conducted tear-down examinations of aircraft components indicate that fatigue crack behavior can be represented conveniently by the following three analysis scales: small three-dimensional cracks at the microscale level, through-the-thickness two-dimensional cracks at the local structural level, and long cracks at the global structural level. The computational requirements for each of these three analysis scales are described in this paper.

Introduction

The ability to predict analytically the onset of widespread fatigue damage in fuselage structures requires methodologies that predict fatigue crack initiation, crack growth, and residual strength. Mechanics-based analysis methodologies are highly desirable because differences in aircraft service histories can be addressed explicitly and rigorously by analyzing different types of aircraft and specific aircraft within a given type. Each aircraft manufacturer has developed mature in-house durability and damage-tolerance design and analysis methodologies that are based on their product development history. To enhance these existing successful methodologies, NASA has adopted the concept of developing an analytical "tool box" that includes a number of advanced structural analysis computer codes which, taken together, represent the comprehensive fracture mechanics capability required to predict the onset of widespread fatigue damage. These structural analysis tools have complementary and specialized capabilities ranging from a nonlinear finite-element-based stress-analysis code for two- and three-dimensional built-up structures with cracks to a fatigue and fracture analysis code that uses stress-intensity factors and material-property data found in "look-up" tables or from equations. The development of these advanced structural analysis methodologies has been guided by the physical evidence of the fatigue process assembled from detailed tear-down examinations of actual aircraft structure. In addition, critical experiments are being conducted to verify the predictive capability of these codes and to provide the basis for any further methodology refinements that may be required. These experiments are essential for analytical methods development and verification, but represent only a first step in the technology-validation and industry-acceptance processes. Each industry user of this advanced methodology must conduct an assessment of the technology, conduct an independent verification, and determine the appropriate integration of the new structural analysis methodologies into their existing in-house practices. NASA has established cooperative programs with U.S. aircraft manufacturers to facilitate this comprehensive transfer of this technology by making these advanced methodologies available to industry.

This paper presents the analytical framework for predicting the onset of widespread fatigue damage. After a discussion of the results of a thorough tear-down fractographic inspection of a five-bay fuselage lap splice joint containing widespread fatigue damage, the analytical fracture mechanics requirements to predict crack initiation, fatigue crack growth, and residual strength are presented. For each of these three fracture mechanics scales, example calculations will be compared to the results of experimental verification tests.

Fractography of Widespread Fatigue Damage (WFD) in a Structural Fatigue Test Article

Valid analytical methodology to predict the onset of widespread fatigue damage in fuselage structure must be based on actual observations of the physical behavior of crack initiation, crack growth, and fracture. The methodology presented herein is based largely on the results of teardown fractographic examinations of aircraft fuselage components. A large section of a fuselage containing a longitudinal lap splice joint extending for five bays was provided to NASA by an aircraft manufacturer after conducting a full scale fatigue test [1]. A photograph of the panel along with a schematic is shown in Figure 1. The fatigue test was terminated after reaching the number of fuselage pressurization cycles that equaled approximately three times the orignal economic design life goal of the aircraft established by the manufacturer. This section of the fuselage was selected because visual inspections made during the test had detected the growth of fatigue cracks extending from adjacent rivets which eventually linked up to form a long crack that extended completely across the bay. Further visual examinations of this section of the fuselage after completing the full scale fatigue test suggested that this section contained widespread fatigue damage. All rivet holes in each of the five bays of the panel were microscopically examined for fatigue cracks. The results of this examination form the physical basis for the analytical methodology develop by NASA to predict the onset of widespread fatigue damage.

There were three principal objectives of the fractographic examination of the fuselage panel. The first objective was to characterize widespread fatigue damage in a fuselage splice joint by assembling a database on the initiation and growth of fatigue cracks from rivets, including identifying the initiation mechanisms. The second objective was to provide a basis for comparing the crack growth behavior simulated in laboratory test specimens to the real behavior of an actual aircraft component. The third objective was to serve as a benchmark to verify the predictive capability of the fatigue crack growth portions of the widespread fatigue damage analytical methodology. This latter objective was achievable because the loading history of the full scale fatigue test article was fully documented. Also, periodic underloads during the fatigue test were used to establish marker bands on the fatigue crack surfaces.

Achieving the above three objectives resulted in the development of a very large database. The contents of this database include:

- maps of cracks as a function of rivet locations in five consecutive bays,
- documentation of crack growth shapes and dimensions,
- identification of the crack initiation location and the initiating mechanisms such as high local stress, fretting, and manufacturing defects,
- analysis of fatigue marker bands,
- correlation between cycles and crack growth behavior, and
- indications of out-of-plane displacements and mixed-mode fracture behavior.

In addition to the extensive database assembled from the teardown fractographic examinations of the panel, sections of other retired aircraft and full scale fatigue articles have also been examined to insure that the analytical methodology under development is sufficiently comprehensive to represent all fuselage assembly practices and design details.

Several general conclusions are obvious from the database. First, fatigue cracks were present at virtually every rivet hole in the top row of rivets. The cracks ranged in size from about 50 microns to several inches. Crack initiation mechanisms included high local stresses, fretting along mating surfaces, and manufacturing defects created during the riveting process. The cracking behavior in each bay was similar and the results of the fatigue marker bands were relatively independent of rivet hole location. An example of small cracks found in the panel is shown in Figure 2. A small crack initiating due to high local stresses within the rivet countersunk hole is shown in Figure 2 (a). The accompanying schematic shows the location of the crack in the outer skin of the lap splice joint. An example of a small crack initiating due to fretting is shown in Figure 2 (b) along with a schematic showing the interface where the fretting occurred. Examples of long cracks found at rivet holes are shown in Figure 3. Figure 3 (a, b, and c) shows a crack that initiated due to high local stresses and Figure 3 (d, e, and f) shows a crack in a different rivet hole that initiated by fretting. The higher magnifications of the cracks shown in Figures 3 (c and f) help to identify the location where the crack initiated and the initiation mechanism. Referring to Figure 3 (b), it is seen that the crack has grown completely through the thickness of the outer skin and has extended a considerable distance beyond the head of the rivet. Likewise, the crack shown in Figure 3 (e) has also extended a considerable distance from the head of the rivet but has not broken through the outer surface of the skin. In both cases, the crack front is curved and indicates the existence of significant bending stresses across the lap splice joint. Figure 4 shows a large crack that has been formed by the link up of the small fatigue cracks that developed at adjacent rivet holes. As can be seen in the photograph, the crack extended into the tear strap region, changed crack growth directions, and grew into a rivet hole in the tear strap. The surfaces of the individual fatigue cracks between the rivets were clearly identifiable in the fractographic examination of the long crack surface. Close examination revealed several cracks that initiated due to high local stresses and other cracks that initiated due to fretting. However, the length of all of the fatigue cracks at link up were approximately the same. This observation suggests that the long crack behavior is somewhat independent of the initiating mechanism. Furthermore, this observation and the quantitative data obtained from the marker band analyses strongly suggest that the fatigue behavior of the long cracks is deterministic and predictable.

Analytical Framework for the Methodology to Predict the Onset of Widespread Fatigue Damage

It is obvious from the above described fractographic examinations that the computational capability required to predict analytically the onset of widespread fatigue damage must be able to represent a wide range of crack sizes from the material (microscale) level to the global structural-scale level. These studies indicate that the fatigue crack behavior in aircraft structure can be represented conveniently by the following three analysis scales: small three-dimensional crack geometry's at the microscale level **(Scale I: Crack Initiation**); through-the-thickness two-dimensional crack geometry's at the local structural level (**Scale II: Fatigue Crack Growth**); and long cracks at the global structural level **(Scale III: Residual Strength**). The computational requirements for each of these three analysis scales are described in the following paragraphs.

Scale I: Crack Initiation

The first analysis scale and corresponding computational capability represents the fracture mechanics of small cracks that exhibit three-dimensional crack-growth behavior. The existence and growth of these small cracks do not affect the global structural deformation states or internal load distributions. Examples of these cracks are surface and corner cracks that initiate at the edges of plates or at holes.

Criterion for Crack Initiation Based on Small Crack Behavior

Small fatigue cracks in some materials grow faster and at lower stress-intensity factor levels than is predicted from large-crack data which exhibits an apparent threshold for crack growth [2]. The initiation and growth of these small cracks are affected by metallurgical features such as inclusion particles and grain-boundary interactions [3]. Typical large-crack results are obtained from tests with cracks greater than about 2 mm in length. The large-crack threshold is usually obtained from load-reduction tests. Some tests and analyses have shown that the development of the threshold may be caused by an increase in crack-closure behavior as the load is reduced. Small cracks that initiate at inclusion particles, voids, or weak grains do not have any prior plastic deformation to develop crack closure. If a small crack is fully open, then the stress-intensity factor range is fully effective and the crack growth rate for the small crack is faster than the rate exhibited by the large-crack data and at a lower stress-intensity factor range.

The concept of crack closure to explain crack growth acceleration and retardation was pioneered at NASA Langley almost two decades ago [4]. The closure concept is based on the postulate that the wake of plastically deformed material behind an advancing crack front may prevent the crack from being fully open during the complete loading cycle. Therefore, only part of the load cycle is effective in growing the crack. The crack closure concept has also been successfully used to explain the small-crack phenomenon exhibited by many aluminum alloys. The successful coupling of the closure methodology with the small-crack growth rate data base has resulted in a total life prediction methodology which treats initiation by predicting the growth of micron size cracks initiating at inclusion particles in the sub-grain boundary microstructure [5].

Computational Methodology for Predicting Crack Initiation in Riveted Structure

Stress-intensity-factor solutions are typically obtained from computational procedures such as the finite element analysis method. The ZIP3D computer code [6] has been developed to model three-dimensional crack configurations and to calculate the corresponding stress-intensity factors. This finite element analysis code uses an eight-node element and can be used to analyze stationary and growing cracks under cyclic elastic-plastic conditions, including the effects of crack closure. The FRANC3D code [7] also has solid modeling capabilities for three-dimensional geometry's based on the boundary element method. For those crack configurations and general loading conditions that may occur for various structural components, weight-function solutions are being developed from the numerical results of parametric studies. These weight-function equations are particularly useful because the stress intensity factor solutions can be obtained from a stress analysis of the uncracked structure. Stress intensity factor solutions are currently being generated for cracks that initiate at countersunk rivet holes. Loading conditions include interference-fit stresses, clamp-up stresses, and loads transferred through a rivet. These stress-intensity factor solutions may then be used as input data for the FASTRAN II code [8] to predict fatigue crack growth. The FASTRAN II code is based on the mechanics of plasticity-induced crack closure. The effects of prior loading history on fatigue behavior, such as crack growth retardation and acceleration, are computed on a cycle-by-cycle basis. The code will predict the growth of cracks exhibiting the "small-crack effect" as well as two- and three-dimensional cracks exhibiting the classical Paris law crack growth behavior. The code has been shown to be especially effective for predicting fatigue crack growth behavior in structures subjected to aircraft spectrum loads. The ZIP3D, FRANC3D, and FASTRAN II codes operate efficiently on engineering workstations, and FASTRAN II also operates on personal computers.

Experimental Verification of Crack Initiation Methodology

The small-crack effect and crack-closure analysis model, FASTRAN II, were used to calculate the total fatigue life (S-N) behavior of single-edge notched (SENT) specimens under constant amplitude and spectrum loads using an initial defect size based on microstructural data at initiation sites. Predicted results for aluminum alloy 2024-T3 were made using an initial semi-circular crack size, 0.00024" (6 microns), that had an equal area to the average inclusion-particle sizes that were experimentally observed to initiate cracks [3]. Comparisons of experimental and predicted fatigue lives of the SENT specimens under the TWIST [10], FALSTAFF [11], and Gaussian [12] load sequences are shown in Figure 5. The specimens were cycled until a crack, length 2a, had grown across the full thickness, B. The predicted lives are in very good agreement with the test data.

Scale II: Fatigue Crack Growth

The second analysis scale and corresponding computational capability represent the fracture mechanics of fatigue cracks that extend through the thickness of a skin or stiffener and are no longer three-dimensional in their crack-growth behavior.

Crack Closure Concept for Fatigue Crack Growth

As discussed in the previous section, the plasticity-induced crack closure model has been shown to be quite accurate in predicting the fatigue crack growth in aluminum alloys for a number of basic crack configurations for both constant amplitude and spectrum loads. The closure model is very accurate for a full range of R ratios and spike overload conditions provided the crack growth rate data are correlated with the effective stress-intensity factor range.

Computational Methodology for Fatigue Crack Growth in Riveted Structure

Two-dimensional analyses are typically quite adequate for predicting crack growth. However, accurate modeling of structural details is required to provide high-fidelity results for the local stresses in a structure so that the fracture mechanics calculations will be accurate. The FRANC2D finite element analysis code [13] has been developed for the analysis of two-dimensional planar structures, and the STAGS (STructural Analysis of General Shells) nonlinear shell analysis code [14] has been developed for general shell structures. The FRANC2D code, developed by Cornell University, is a user-friendly engineering analysis code with pre- and post-processing capabilities especially developed for fracture mechanics problems. The code operates on UNIX-based engineering workstations with X-Windows graphics and is interactive and menu driven. A unique capability of the code is the ability to predict non-self-similar crack growth behavior. An automatic adaptive remeshing capability allows an engineer to obtain a history of the stress intensity factors for any number of cracks in the structure and for any arbitrary crack growth trajectory. The STAGS finite element code, developed by Lockheed Palo Alto Research Laboratory, provides the capability to model any general shell structure and has both geometric and material nonlinear analysis capabilities. STAGS is particularly well suited for analyzing shells that have structural features such as frames, stiffeners, and cutouts. The code uses the Riks arc-length projection method and computes large displacements and rotations at the element level. The code has been developed especially for nonlinear stability and strength analyses. Both FRANC2D and STAGS can calculate the history of the stress intensity factors for a growing crack that are compatible with FASTRAN II so that fatigue crack growth analyses may be performed. Other crack growth models may also be used. STAGS and FRANC2D operate on engineering workstations and mainframe computers.

The advanced durability and damage-tolerance analysis capabilities developed in the NASA Airframe Structural Integrity Program will also be implemented in the NASGRO analysis code [15]. NASGRO is a general-purpose damage tolerance analysis code being developed by NASA Johnson Space Center. The code is based on fracture mechanics principles and may be used to compute stress intensity factors, fatigue crack growth, critical crack sizes, and the limit of safe life. An extensive library of stress intensity factors may be used with NASGRO or solutions may be obtained from a boundary element analysis capability using the FADD analysis code [16]. NASGRO also has an extensive material property library which includes most aluminum alloys, titanium alloys, and steels commonly used in the aerospace industry. Fatigue crack growth may be computed from a crack-closure mechanics model or from one of several empirical models commonly used by industry. NASGRO is used extensively throughout the aerospace industry. FADD was developed at the University of Texas and uses the distributed-dislocation method to compute stress intensity factors. This approach combines a highly accurate stress intensity factor analysis with the modeling simplicity of the boundary element analysis method. FADD is also available in a stand-alone version and is currently being tested by industry at beta-site locations. NASGRO operates on engineering workstations and personal computers. FADD is also available as a stand-alone code and operates on personal computers.

Experimental Verification of Fatigue Crack Growth Methodology

As part of the experimental verification of the fatigue crack growth methodology, fatigue tests on single shear riveted lap joint specimens were conducted to measure the growth of cracks from countersunk rivets holes [17]. The specimens were made from 0.040" thick Alclad 2024-T3 sheet material and 5/16" rivets. A small EDM notch was used to initiate the fatigue crack at the rivet hole. The EDM notch was placed in the countersunk hole before the rivet was expanded. Tension-tension fatigue tests (R=0.02) were conducted and the growth of the fatigue crack from the EDM notch was recorded throughout the test. The growth of the through-thickness cracks were recorded along the surface of the specimen using an optical microscope. Special restraints were used to prevent the large out-of-plane rotations caused by the eccentricity in the load path of the single shear specimen. The fatigue tests were terminated usually after appreciable crack growth was recorded. Several specimens were cut apart and the interference fit was estimated by measuring the size of the rivet hole after expanding the rivet relative to the original machined hole size.

The FRANC2D code was used to analyze the specimen and compute the crack growth rates. The analysis included the effects of interference fit stresses, bearing load distribution in the countersunk hole, and an approximation of the stress concentration produced by the three-dimensional geometry of the countersunk hole. The experimental measurements of the interference fit expansion was used to compute the interference fit stresses. The analytical reduction of the data recorded during the fatigue tests is shown in Figure 6. The measured crack growth rates are plotted against the computed stress-intensity factor range. The open symbols are the crack growth test data from the single shear specimens. The solid line is the baseline da/dn crack growth data for aluminum alloy 2024-T3 measured from standard crack growth tests. The length of the cracks growing from the rivet in the lap specimen were measured experimentally and the FRANC2D model was used to calculate the corresponding stress-intensity factors. The coalescence of the data from the single shear lap specimen around the baseline crack growth data confirms the accuracy of the FRANC2D analysis to properly account for the complex stress state at the countersunk rivet. The gray shaded area shows the values of the crack data if the stress intensity factors were computed without including interference fit stresses. Without the interference fit, the cracks grow at a faster rate for the same applied stress. The benefit in the crack growth rate is produced by the tensile residual stresses caused by the interference which results in a reduction to the stress intensity-factor range.

Scale III: Residual Strength

The third analysis scale and corresponding computational capability represent structures with long cracks that change the internal structural load distribution, that exhibit behavior strongly affected by structural details, and that affect the residual strength of the structure. In addition, the fracture mechanics of ductile materials such as 2024-T3 aluminum alloy often requires an elastic-plastic stress analysis capability that predicts stable tearing and fracture. Furthermore, nonlinear geometric effects, such as crack bulging in shell structures, also significantly affect residual strength predictions. All of these complexities are present in a fuselage shell structure and must be represented in a residual-strength analysis of the fuselage.

The structural analysis computer codes under development in the NASA Airframe Structural Integrity Program are being integrated into an analytical methodology for predicting the residual strength of a fuselage structure with one or more cracks. The analytical prediction of the residual strength of a complex built-up shell structure, such as a fuselage, requires the integration of a ductile fracture criterion, a fracture-mechanics analysis, and a detailed stress analysis of the structure. The crack-tip opening-angle (CTOA) criterion has been experimentally verified to be a valid fracture criterion for mode I stress states in thin and moderately thick (0.5-inch-thick or less) aluminum alloys. The CTOA criterion has been demonstrated to be valid for predicting the link-up of a long lead crack with small fatigue cracks ahead of the advancing lead crack. This fracture criterion has been implemented into the STAGS geometric and material nonlinear finite-element-based shell analysis code to provide an integrated structural-integrity analysis methodology. The capability to model a growing crack that may extend in a non-self-similar direction has been added to the STAGS code along with an automated mesh refinement and adaptive remeshing procedure. The topological description of the growing crack is provided by the FRANC3D fracture mechanics code. The geometric nonlinear behavior of a stiffened fuselage shell is currently under study for internal pressure loads combined with fuselage body loads that produce tension, compression and shear loads in the shell.

The CTOA Fracture Criterion for Residual Strength

The critical crack-tip opening-angle (CTOA), or equivalently, the crack-tip opening-displacement (CTOD), fracture criterion is a "local" approach to characterizing fracture. In contrast, the J-integral or J-R curve criterion is based on global deformations and has been found to be specimen and crack-size dependent for structures with large amounts of stable tearing. The constant CTOA (or CTOD) criterion has been used to predict the variations in J-R curves due to differences in crack sizes and specimen types. Therefore, a local crack-tip displacement is a more fundamental fracture parameter than the J-integral representation for local strain-controlled fracture processes such as stable tearing and void coalescence.

Simple plastic-zone models that are based on linear-elastic stress intensity factors can be adjusted to fit experimental data and then used to predict crack link-up for relatively simple structural geometry's. While these methods predict the correct trends in crack link-up behavior, they may be difficult to apply to analyses of complex structural details that are characteristic of a fuselage structure. The CTOA criterion can be effectively implemented into a finite element analysis code provided that the code has elastic-plastic deformation and crack-growth simulation capabilities. These capabilities exist in the STAGS geometric and material nonlinear shell analysis code, but analyses of large-scale problems must currently be conducted on a high-performance mainframe computer. After thorough experimental verification of the residual strength analysis methodology, it is anticipated that the methodology can be simplified by taking advantage of appropriate engineering approximations.

An extensive test program [18] has been conducted to interrogate experimentally the characteristics of the CTOA criterion and to establish its validity as a fracture criterion for thin-sheet aluminum alloy 2024-T3. A schematic of the four basic flat-panel geometry's used to verify the elastic-plastic finite-element code and the CTOA criterion for mode I fracture is shown in Figure 7. The blunt-notch panel was used to verify the finite element analysis code used to compute plastic deformation fields and large displacements. Measurements of far-field displacements and the local displacements inside the open holes at the ends of the crack were accurately predicted by the finite element analysis for large-scale plastic deformations. The center-crack and three-hole-crack panels were used to measure the load (or far-field applied stress) as a function of crack extension and the CTOA during stable tearing. Because the tests were conducted at a specified controlled displacement rate, crack extension was measured well beyond the maximum load observed during the test. Stable tearing was quite extensive in the three-hole-crack specimen because the crack driving force is reduced as the crack approaches the two large open holes in a manner that is similar to the behavior of cracks in stiffened panels. A high-resolution long-focal-length microscope was used to record the stable-tearing results. The microscope image was videotaped, digitized, and recorded in a computer file. The tearing event was then analyzed on a frame-by-frame basis and the critical opening angle was measured throughout the fracture event. A typical CTOA measurement is shown in Figure 8. As can be seen in the figure, the opening angle is relatively insensitive to the length over which the angle is measured. The results of a three-hole-crack panel test are given in Figure 9. After an initial transition region, the CTOA is constant throughout the stable-tearing process. The initial transition region is caused by a three-dimensional effect that occurs as the crack tunnels and transitions from flat- to slant-crack growth. Over 63 mm (2.5 in.) of stable tearing was recorded and the CTOA values were nearly constant. Measurements such as these were also made for center-crack and three-hole-crack panels of various widths, crack lengths, and sheet thicknesses ranging from 1.0 mm (0.04 in.) to 2.3 mm (0.09 in.). Also, measurements of the CTOA were obtained for compact tension specimens. In all cases, the measured CTOA was approximately 6.0-degrees for cracks oriented in the LT direction of the sheet and 5.1-degrees for cracks oriented in the TL direction, where L designates the principal rolling direction of the sheet and T designates the direction transverse to the principal rolling direction. A complete description of these test results is given in reference 18.

A series of fracture tests were conducted by the National Institute of Standards and Technology (NIST) [19] on aluminum panels to characterize the fracture behavior and link-up of multiple cracks in very wide panels. Ten flat panel test specimens at 3988 mm (157.0 in.) long, 2286 mm (90.0 in.) wide, and 1.026 mm (0.040 in.) thick were fabricated from single sheets of bare 2024-T3 aluminum alloy. Three center-crack panels with a single long center crack and seven panels with a long crack and small multiple-site damage (MSD) cracks ahead of the single long crack were tested to failure. Saw cuts were used to simulate fatigue cracks. Specially designed grips and anti-buckling guides were used to conduct the tests in the 1780-kN capacity universal testing machine at NIST. (One test , MSD #6, was conducted without anti-buckling guides and the results from this test will be discussed later.) The load and displacement histories were recorded for each test and the fracture events were recorded on film, video tape, computer, magnetic tape, and occasionally optical microscopy. The first three tests were used to measure basic material fracture properties such as the R-curve and critical crack-tip opening angle (CTOA). The other six tests, with anti-buckling guides, were link-up and fracture tests of panels with various multiple-site damage crack configurations.

The CTOA fracture criterion was used to analytically predict the fracture of the wide panels with MSD cracks. The center-crack panels with the single long crack were used to determine the value of the CTOA to be used in the MSD analyses. In order to match the load, displacement, and crack extension data recorded during the stable tearing and fracture of the single crack panels, a critical CTOA value of 3.4 degrees had to be used. In addition, an initial crack opening displacement of 0.0086 inches also had to be used in the analysis to properly simulate the initial stable tearing from the saw cuts. Using these values of CTOA and initial displacement, the applied load to crack link-up and final fracture were calculated for each MSD test prior to conducting the test. The crack configurations, experimental test loads at panel fracture, and the analytical predictions using the CTOA criterion are given in Table I. As can be seen the predictions are within + or - 6% in all test but one which was off by 11%. It is interesting to note that MSD test number 10 is a repeat of MSD test number 7, with the identical crack configuration. This test is the only MSD test with multiple experimental failure loads. Note that the experimental failure loads varied by about 10%. Therefore, the analytical predictions are viewed to be within the experimental accuracy of the tests data.

It should be noted that the CTOA angle experimentally measured during the fracture tests was consistently about 5.5 degrees. This value is significantly higher than the value of 3.4 degrees required to analytically simulate the single-crack fracture test behavior. This difference is believed to be attributed to the ineffectiveness of the anti-buckling guide to prevent out-of-plane displacements during the fracture tests. Visual observations made during the tests suggested that the panels did buckle even with the anti-buckling guides. An additional test, MSD #6, was conducted without anti-buckling guides to determine quantitatively the effect of panel buckling on the fracture load. The test resulted in about a 10% lower failure load than for the panel with the ineffective anti-buckling plates. Subsequent fracture analyses conducted by NASA using the STAGS code and the CTOA fracture criterion have confirmed that the effects of flat panel buckling can result in the magnitude of the discrepancy between the measured CTOA and the value required in the analysis to predict the fracture test results.

The experimental and analytical results presented herein verify the CTOA fracture criterion for predicting the residual strength of flat panels with cracks undergoing mode I fracture behavior. Further testing is required to verify the criterion for predicting the residual strength of complex stiffened shell structures. The CTOA criterion must be extended to mixed-mode loading conditions. Also, numerical procedures for crack extension under mixed-mode loading conditions must be implemented into an elastic-plastic shell analysis code. And finally, the ability to predict crack trajectories accurately and to model curved crack growth must be developed. The next section describes the stiffened shell structural analysis methodology being developed for analyzing a fuselage structure and for predicting its residual strength accurately.

Computational Methodology for Residual Strength of Fuselage Structure

Unique capability has been developed that integrates the fracture topology modeling capabilities of FRANC3D with the general shell analysis capabilities of STAGS into an integrated FRANC3D/STAGS analysis procedure [20]. The automatic adaptive remeshing capability of FRANC3D and the geometric nonlinear stress-analysis capability of STAGS provides the analysis basis required to predict the crack-growth, crack-turning and crack-arrest behavior exhibited by pressurized shell structures in damage-tolerance tests. This capability is described in greater detail in the following paragraphs. For simple two-dimensional plane-stress or plane-strain fracture mechanics problems, the ZIP2D special-purpose finite element code [21] has proven to be very accurate and computationally efficient. The integrated FRANC3D/STAGS analysis procedure currently operates on high-level workstations or on mainframe computers, and ZIP2D operates on workstations.

The STAGS nonlinear finite element analysis code has been modified to include the capability of conducting crack growth and residual strength analyses for stiffened fuselage shell structures subjected to combined internal pressure and mechanical loads. STAGS was originally developed to predict the strength, stability and nonlinear response of non-axisymmetric or general shells and includes analyses for both geometric and material nonlinear behavior. The nonlinear solution algorithm used in STAGS is based on Newton's method and includes both the modified and full versions of Newton's method. Large rotations are represented by a co-rotational algorithm at the element level, and the Riks arc-length projection method is used to integrate past limit points. The finite element library includes nonlinear beam, plate, and shell elements. Complex stiffened shell structures can be modeled to include as many finite elements as required to represent accurately the response of each structural member in the stiffened shell of interest. The computational efficiency of the code allows nonlinear analyses of models with over 100,000 degrees of freedom to be conducted in a reasonable amount of computer time. Both self-similar and non-self-similar crack-growth prediction capabilities have been added to STAGS for predicting crack growth in a shell that is in a nonlinear equilibrium state. The crack-growth analysis used in FRANC3D/STAGS is based on a virtual crack extension analysis that calculates the strain energy release rate for nonlinear shells with mixed-mode crack growth including shell wall bending. A load relaxation capability is used to represent the local load redistribution that occurs as a crack grows in the shell and Newton's method is used to maintain nonlinear equilibrium as the crack propagates. Nonlinear adaptive mesh refinement is used to determine the necessary finite element model changes as the crack propagates.

The general strategy for developing the nonlinear structural analysis methodology for predicting residual strength of stiffened shells with cracks is shown in Figure 10. Large-scale global models of a stiffened fuselage shell of interest are developed and nonlinear analyses are conducted to determine the internal load distribution and general response of the shell as shown in the upper left of the figure. A hierarchical modeling approach is used to provide more highly refined local models which are developed based on the global model results. The local models provide the higher-fidelity solutions that are necessary to predict stress and displacement gradients near the crack discontinuity in the shell as shown in the upper right of the figure. Several local models are generated as required and analyzed to provide the detailed stress and deflection results necessary to predict crack growth and residual strength for any structural detail feature such as the longitudinal lap splice shown in the lower right of the figure.

An example [22] of the hierarchical modeling strategy for nonlinear stiffened shell analysis using STAGS is shown in Figure 11. The nonlinear hoop stress and radial deflection results for the global shell model of a frame and stringer stiffened aluminum shell are shown on the left of the figure. The shell has a longitudinal crack at the top of the fuselage and is loaded by 55.2 KPa (8 psi) of internal pressure. The longitudinal crack in the skin is next to a stiffener and the frame at the crack location is also broken. A curved stiffened panel model was developed with five frames and five stringers to generate the 36-skin-bay local model as shown in the upper right of the figure. This model provides more detailed stress- and deflection-gradient results near the cracked region as shown in the figure. The results shown are for a 0.508 m (20.0 in.) long skin crack with the center of the crack at the broken frame. The frames are located at the dark circumferential regions in the figure. The boundary conditions for this local model are based on the results of the global model analysis, and both equilibrium and compatibility with the nonlinear global shell solution are maintained at the panel boundaries. A more refined stiffened panel model was developed with two frames and three stringers to generate the six-skin-bay local model shown in the lower right of the figure. The hoop stress and radial deflection results shown are for a 1.016-m (40.0-in.) long crack that has grown to the frames on either side of the broken frame. The boundary conditions for this more refined local model are based on the results of the 36-skin-bay stiffened panel model and both equilibrium and compatibility with the nonlinear 36-skin-bay panel solution are maintained at the six-skin-bay panel boundaries. This hierarchical modeling and analysis approach provides the high-fidelity nonlinear stress- and deflection-gradient results needed to represent the shell behavior near the crack to the level of accuracy required to predict crack growth and residual strength accurately.

An example of the fracture mechanics analysis capability using FRANC3D/STAGS is shown in Figures 12 and 13. Using the initial STAGS finite element model shown in Figure 12, stable tearing is simulated for a 6 inch, 8 inch, and 10-inch long skin crack centered over a broken tear strap. The FRANC3D fracture analysis is performed using the local two-bay by two-bay geometry model with the edge displacements determined from an intermediate six-bay by six-bay model. As illustrated schematically in both Figures 12 and 13, the boundary conditions for the local and intermediate models were determined from the full-scale fuselage cylinder model shown in Figure 11. The full-scale fuselage model was subjected to an internal pressure of 55.2 Kpa (8 psi) plus shear and bending loads simulating down forces acting on the empennage. The stress contour plot in the upper right of Figure 13 shows the hoop stress resultants for the initial crack length of 6 inches as modeled by the finite element mesh shown in Figure 12. Using the FRANC3D adaptive remeshing capability, the crack is extended to 8.0 inches and then to 10 inches as shown in the two lower stress contour plots in Figure 13. The direction of crack extension was determined by a FRANC3D algorithm using a simple maximum principle stress criterion. Then the new finite element meshes for the 8 inch and 10 inch crack geometry's were automatically developed by the FRANC3D adaptive remeshing algorithms. The crack growth increments for this stable tearing simulation were arbitrarily chosen for illustrative purposes only. As is clearly evident from the stress contour plot, the effect of the shear force results in a non-symmetric and non-self-similar crack extension.

Experimental Verification of Residual Strength Methodology

The methodology to predict analytically the residual strength of a fuselage structure with widespread fatigue damage and discrete source damage will be experimentally verified. NASA will be conducting verification tests of curved stiffened panels simulating sections of the fuselage geometry subjected to internal pressure loads only and subjected to combined loads. Plans call for two tests to be conducted under internal pressure only loads. The first test will be a residual strength test on a panel with a midbay skin crack oriented in the longitudinal direction with the damage located away from a splice joint. The second test will be a residual strength test of a panel with simulated widespread fatigue damage in a longitudinal lap splice joint and with a certification size lead crack simulating discrete source damage. The third and fourth verification tests will be conducted on panels with the same crack configurations as the first two tests but with combined internal pressure, shear, and bending loads simulating fuselage stresses created by the empennage loads. Two special test facilities, a pressure box for internal pressure loads and a D-box for combined internal pressure and mechanical loads, at NASA Langley Research Center will be used to conduct these tests of the built-up shell structures. In addition to the tests to be conducted at Langley, test data available from the industry and other Government funded test programs will also be used as benchmarks to verify the accuracy of the analytical prediction methodology. This experimental verification test program will be completed during the next calendar year.

Summary

A comprehensive analytical methodology has been developed for predicting the onset of widespread fatigue damage in fuselage structure. The determination of the aircraft service life that is related to the onset of widespread fatigue damage includes analyses for crack initiation, fatigue crack growth, and residual strength. Therefore, the computational capability required to predict analytically the onset of widespread fatigue damage must be able to represent a wide range of crack sizes from the material (microscale) level to the global structural-scale level. Carefully conducted tear-down examinations of aircraft components indicate that widespread fatigue damage behavior can be represented by the following three analysis scales: small three-dimensional cracks at the microscale level, through-the-thickness two-dimensional cracks at the local structural level, and long cracks at the global structural level.

A computational "tool box" is being developed that includes a number of advanced structural analysis computer codes which, taken together, represent the comprehensive fracture mechanics capability required to predict the onset of widespread fatigue damage. These structural analysis tools have complementary and specialized capabilities ranging from a nonlinear finite-element-based stress-analysis code for two- and three-dimensional built-up structures with cracks to a fatigue and fracture analysis code that uses stress-intensity factors and material-property data found in "look-up" tables or from equations. The development of these advanced structural analysis methodologies has been guided by the physical evidence of the fatigue process assembled from detailed tear-down examinations of actual aircraft structure. In addition, critical experiments are being conducted to verify the predictive capability of these codes and to provide the basis for any further methodology refinements that may be required. These experiments are essential for analytical methods development and verification, but represent only a first step in the technology-validation and industry-acceptance processes. Each industry user of this advanced methodology must conduct an assessment of the technology, conduct an independent verification, and determine the appropriate integration of the new structural analysis methodologies into their existing in-house practices. NASA has established cooperative programs with the U.S. aircraft manufacturers to facilitate the comprehensive transfer of this advanced technology to industry.

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Table 1. - Comparison of measured and predicted failure loads on NIST MSD fracture tests.

Panel	Number of Sawcuts	Test Load kips	Predicted Load kips	Percent Error
MSD #1	1	77.0	76.8 (a)	-0.3
MSD #2	1	96.3	96.0	-0.3
MSD #3	1	64.9	65.6	+1
MSD #4	7	69.1	67.0	-3
MSD #5	7	91.2	80.9	-11
MSD #7	11	48.2	49.8	+3
MSD #8	21	47.5	50.6	+6
MSD #9	21	79.2	74.8	-6
MSD #10 (b)	11	52.2	49.8	-5

(a) Fitted to test (CTOA = 3.4 deg.; δ_i = 0.0086 in.).

(b) MSD #10 was repeat of MSD #7.

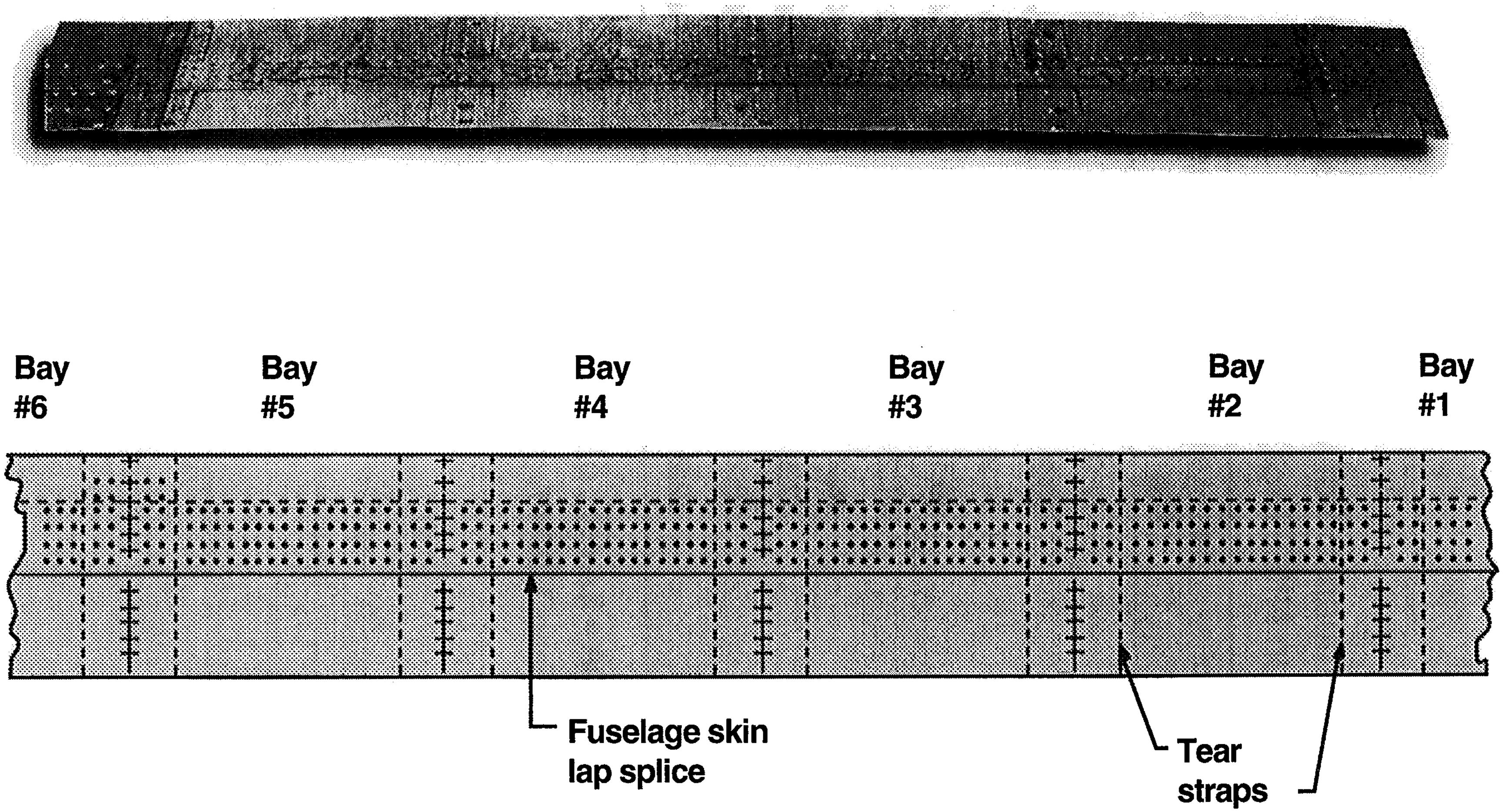

Figure 1. Fractography of WSFD in structural fatigue test article.

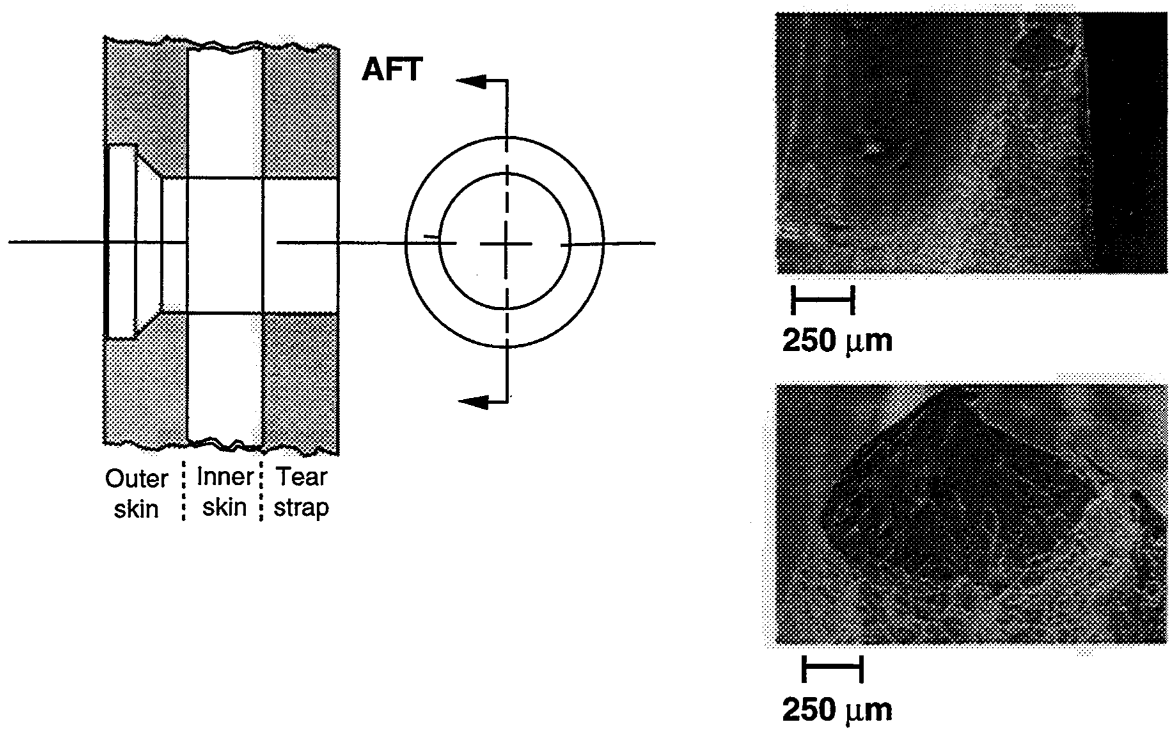

Figure 2(a). Small fatigue cracks at rivets.

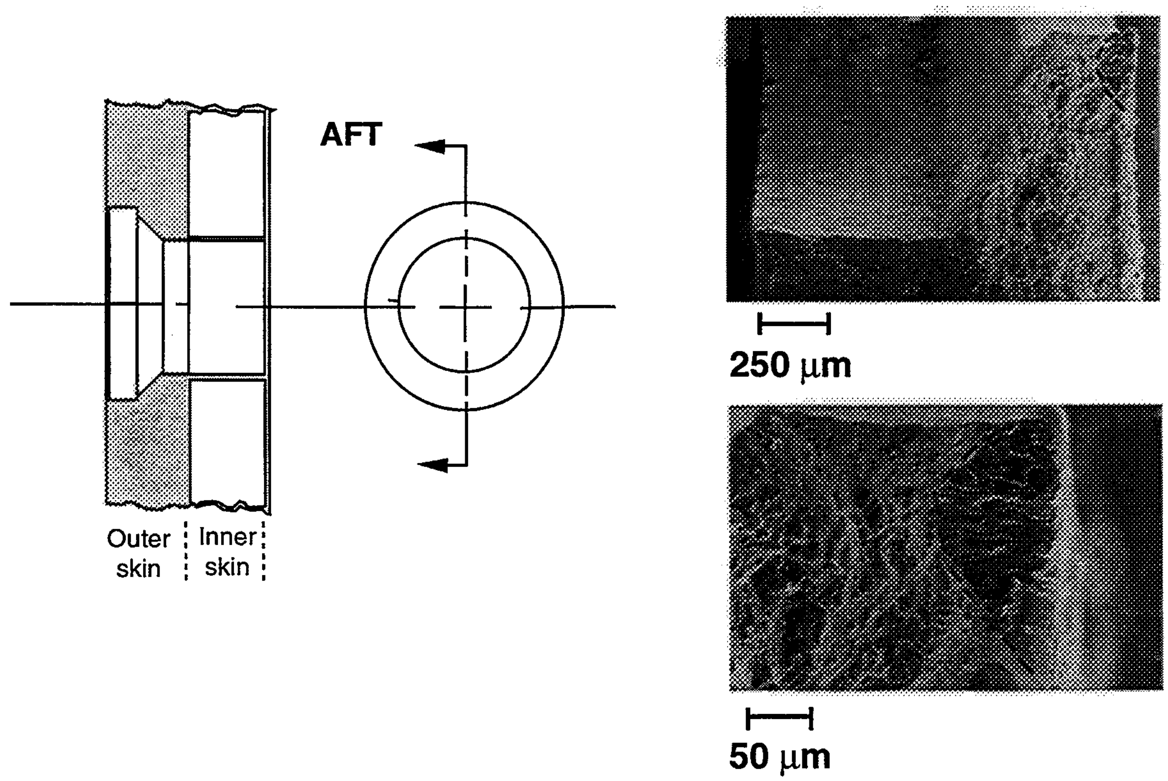

Figure 2(b). Small fatigue cracks at rivets.

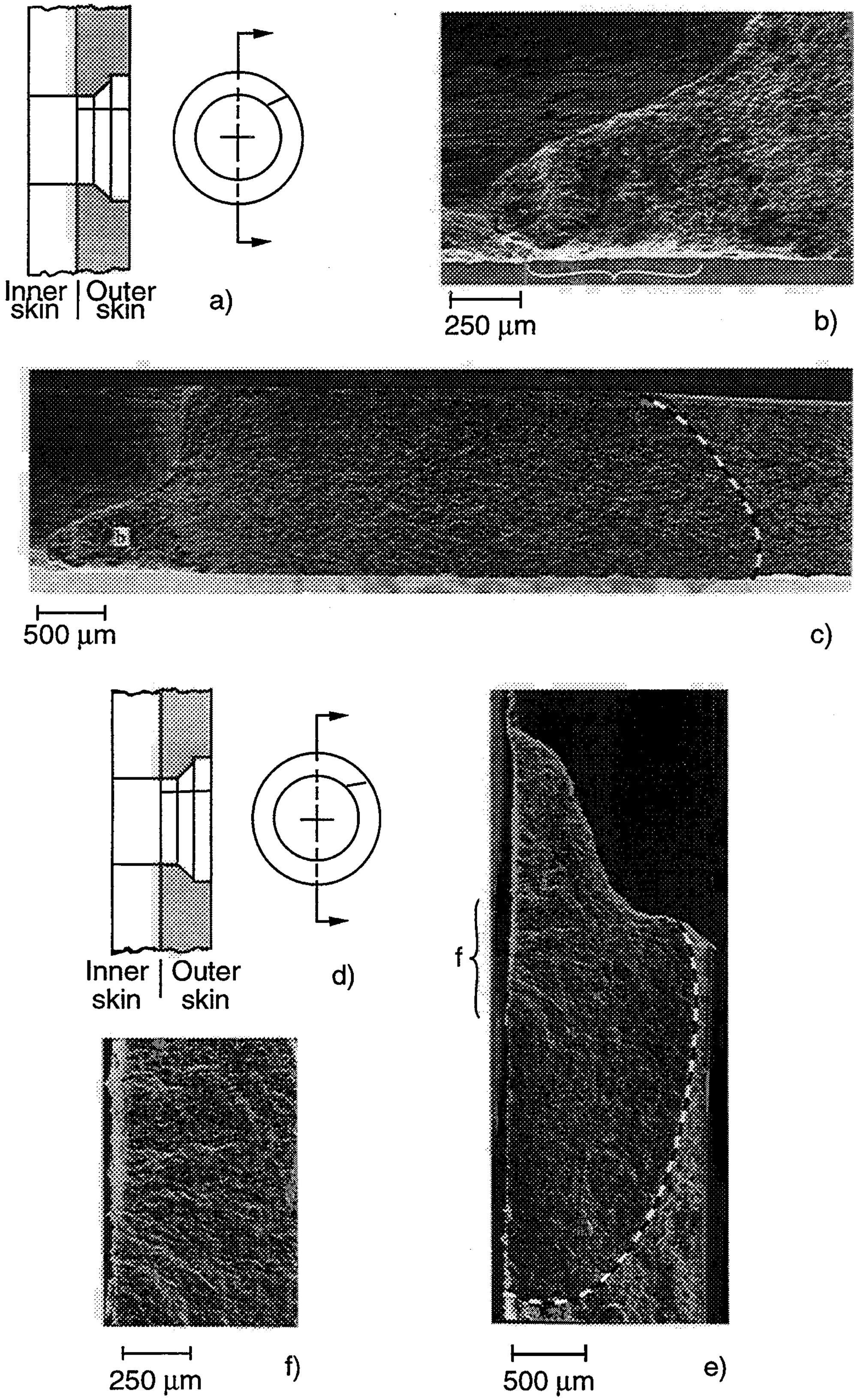

Figure 3. Long cracks at rivets.

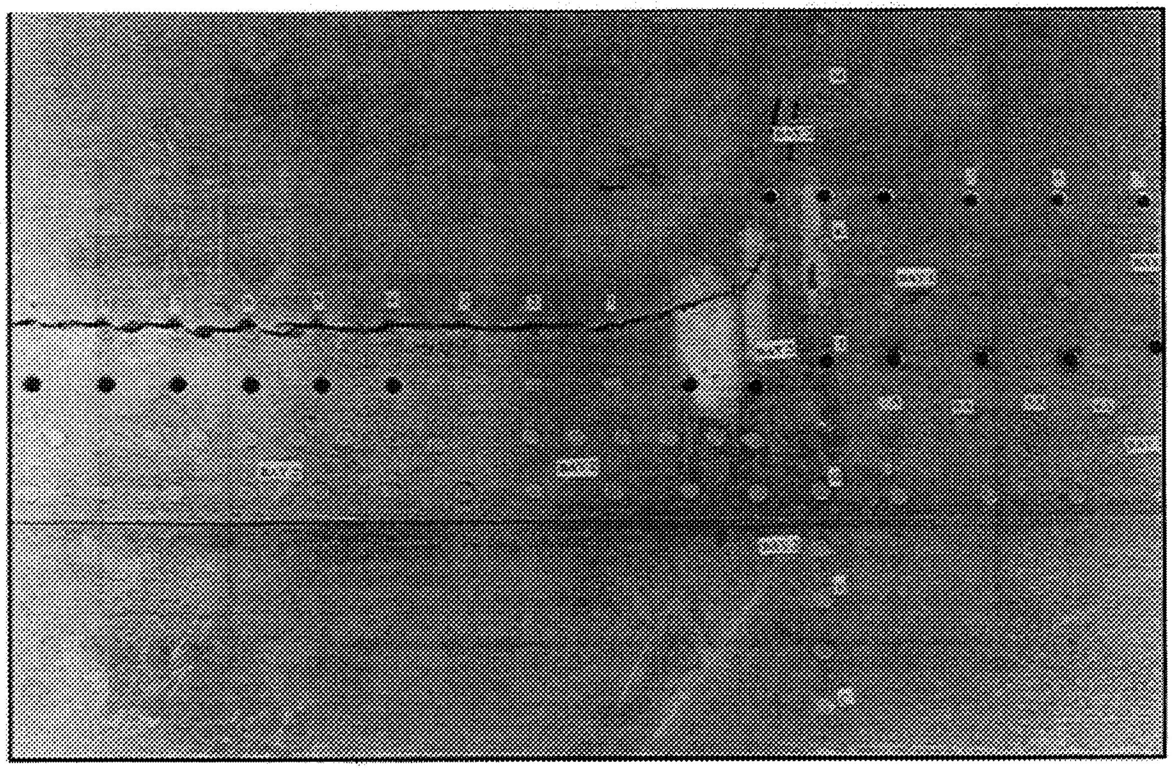

Figure 4. Fatigue crack in a fuselage splice joint of a transport aircraft.

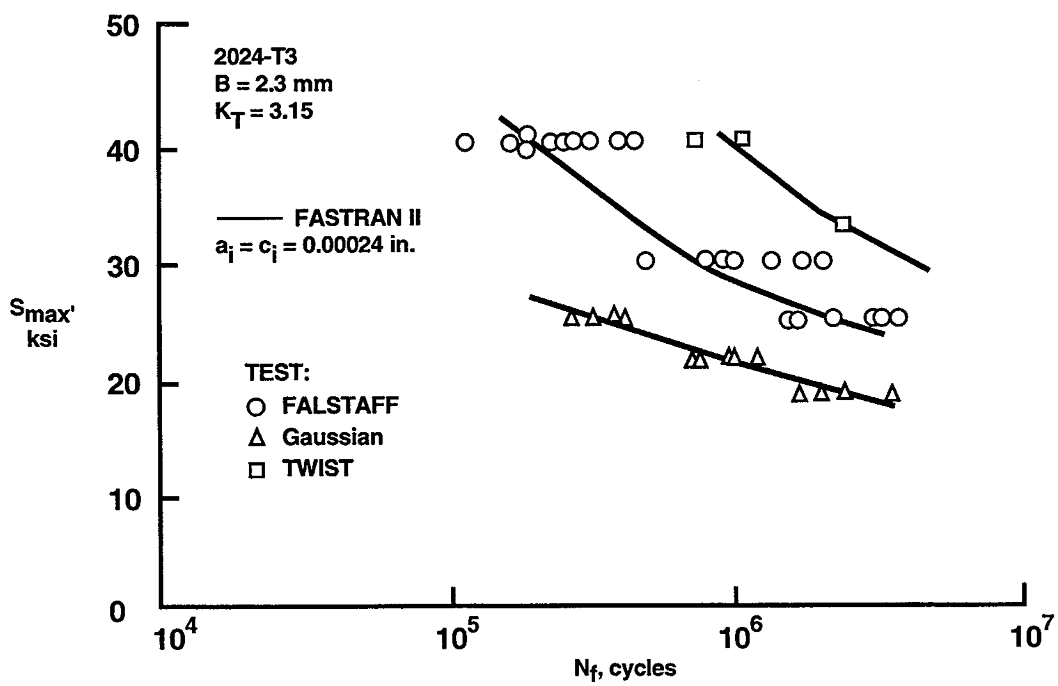

Figure 5. Comparison of test and predicted S-N behavior

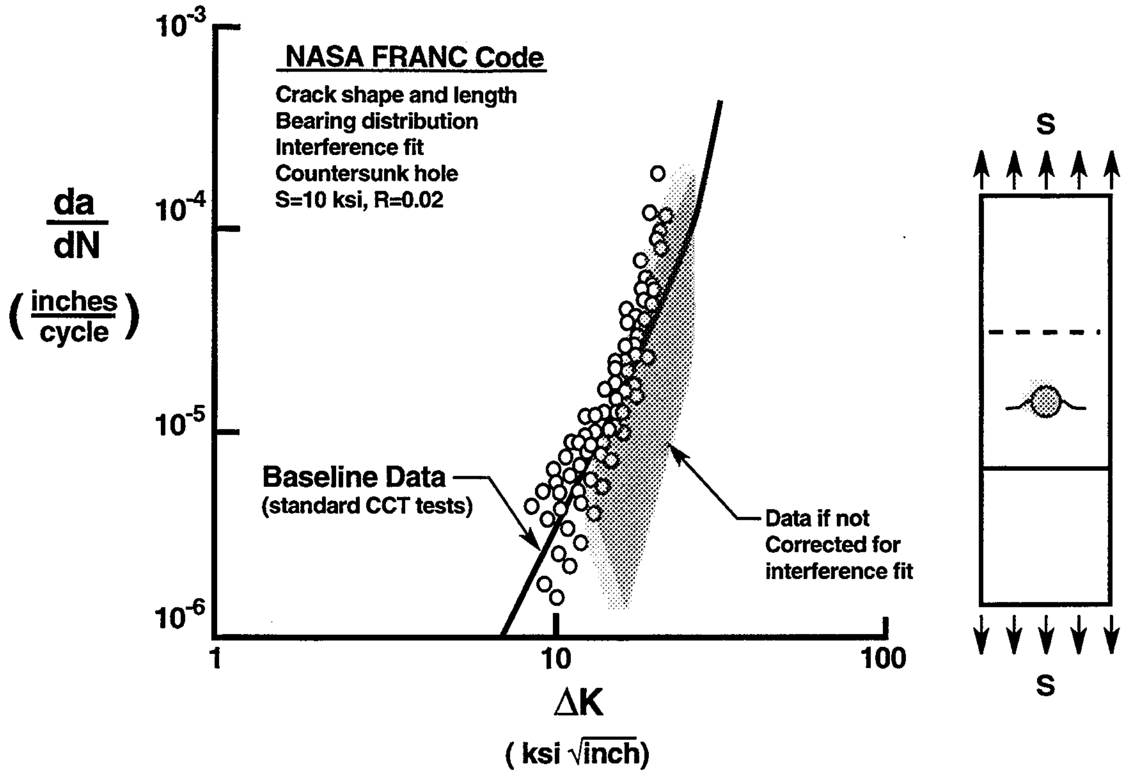

Figure 6. Fracture mechanics of cracks extending from rivets.

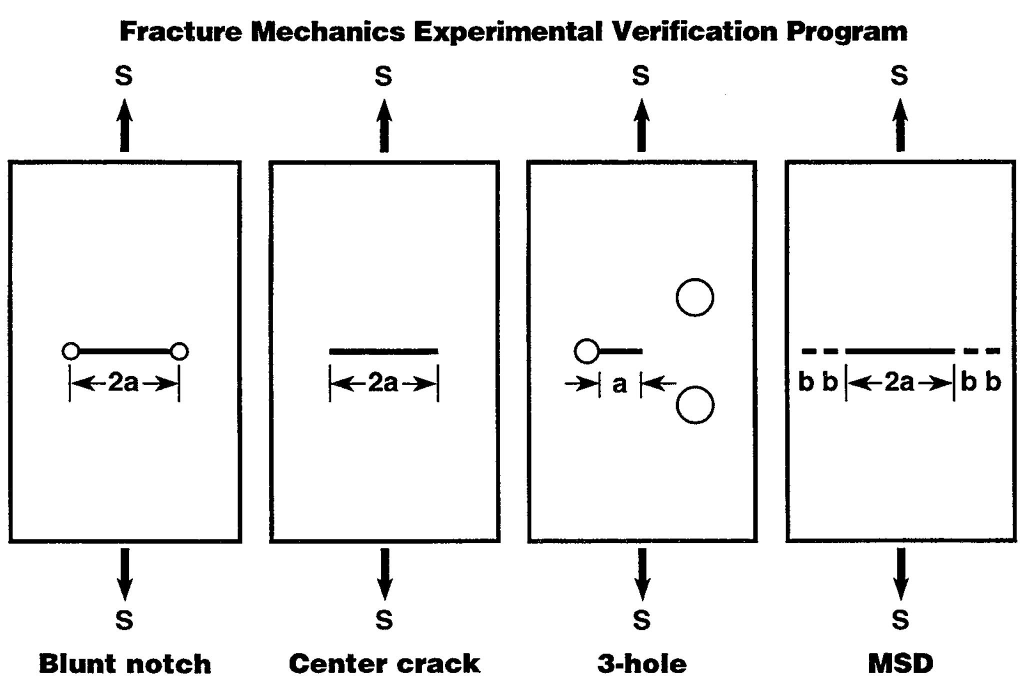

Figure 7. CTOA Fracture criterion for residual strength.

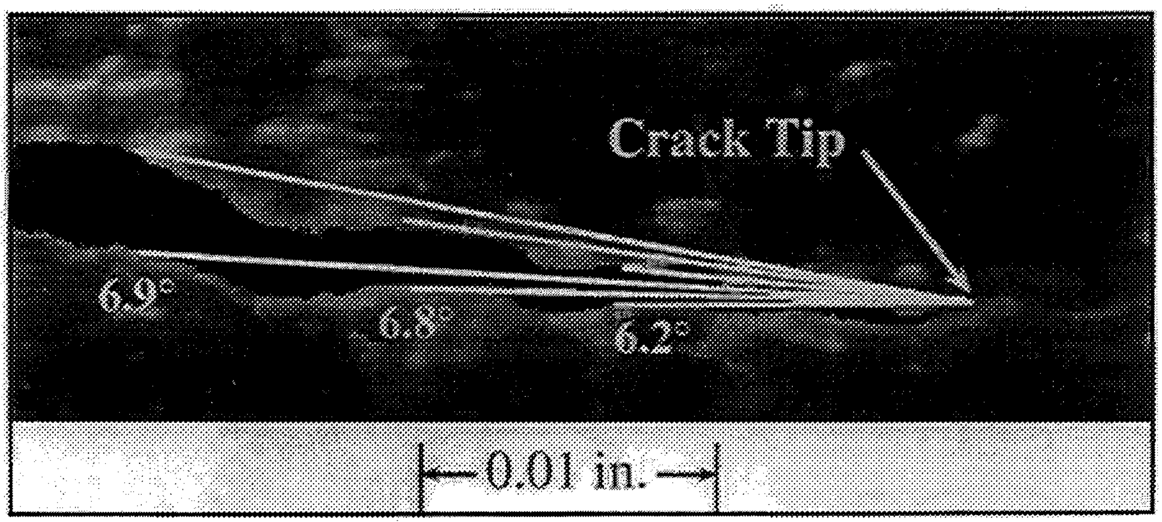

Figure 8. Crack tip opening angle (CTOA) measurements.

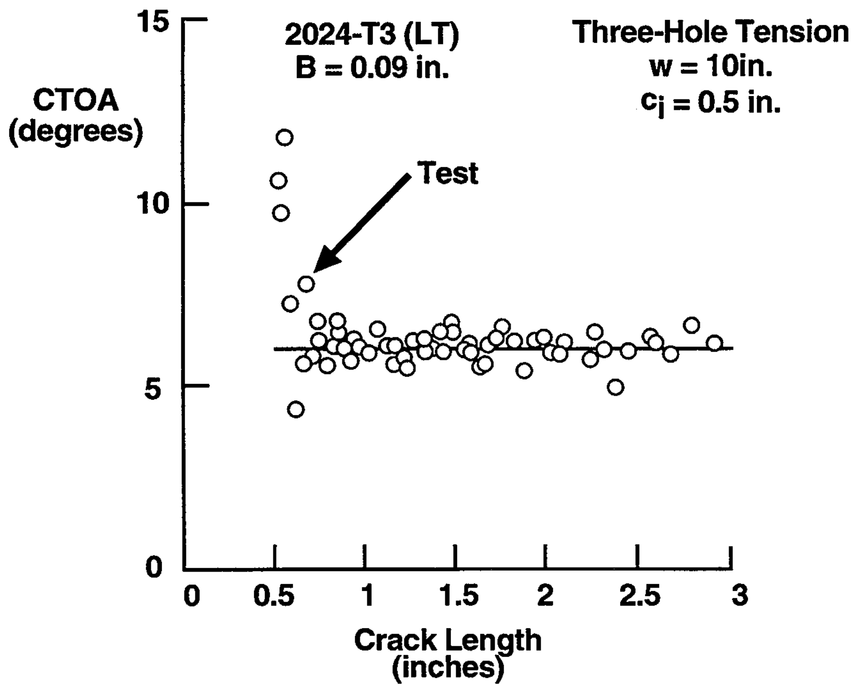

Figure 9. Experimental measurements of CTOA.

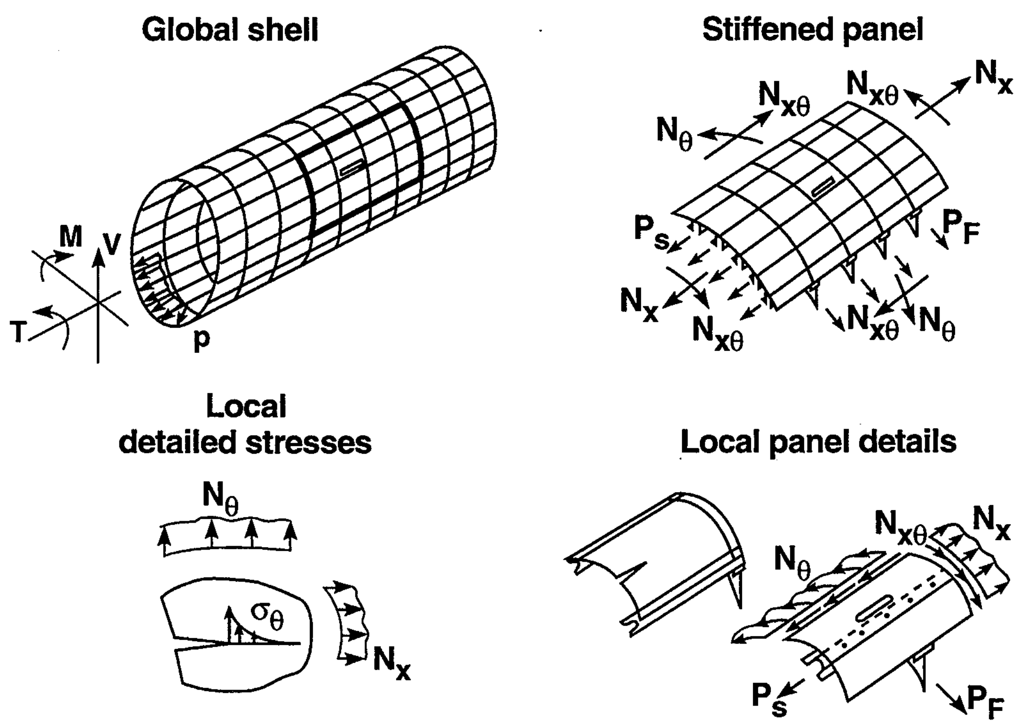

Figure 10. Hierarchical nonlinear stiffened shell models.

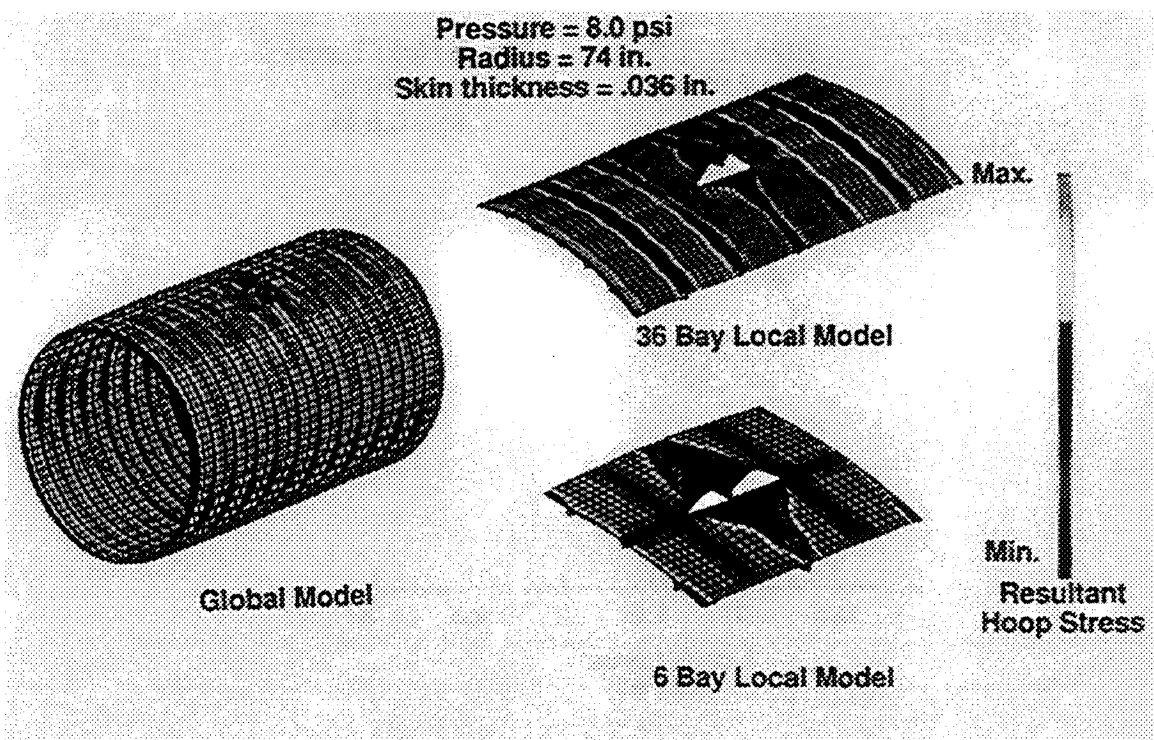

Figure 11. Stiffened aluminum fuselage shell with 20" skin crack and broken frame.

• **Hierarchical models: internal pressure, bending, vertical shear**
• **Local 2-bay by 2-bay model, edge displacements from 6-bay by 6-bay mode**

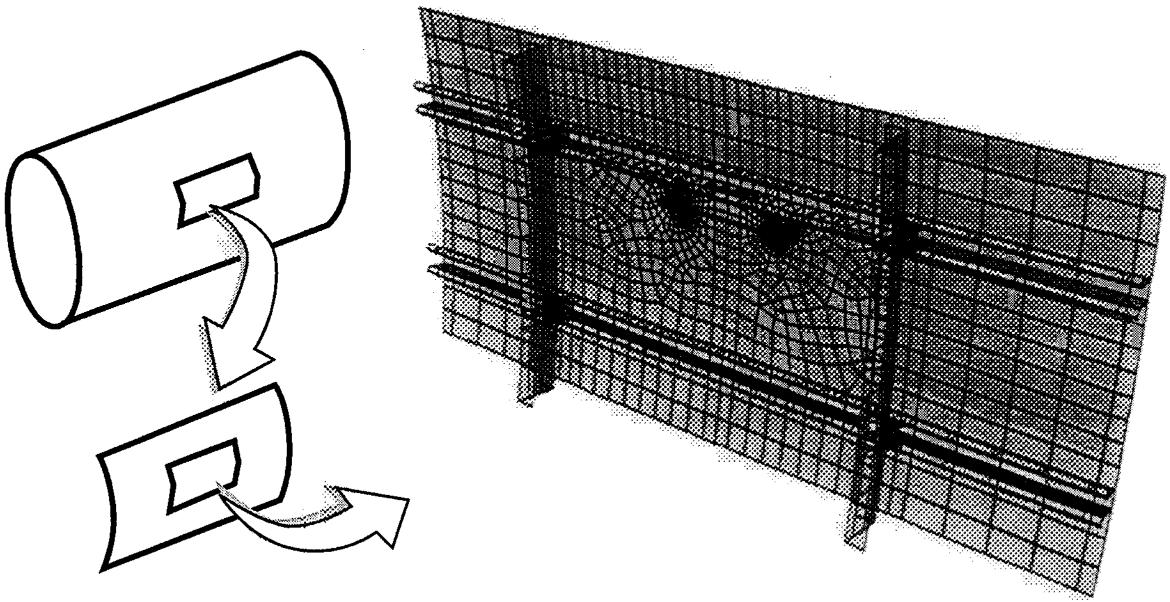

Figure 12. Curvilinear crack analysis using FRANC3D/STAGS.

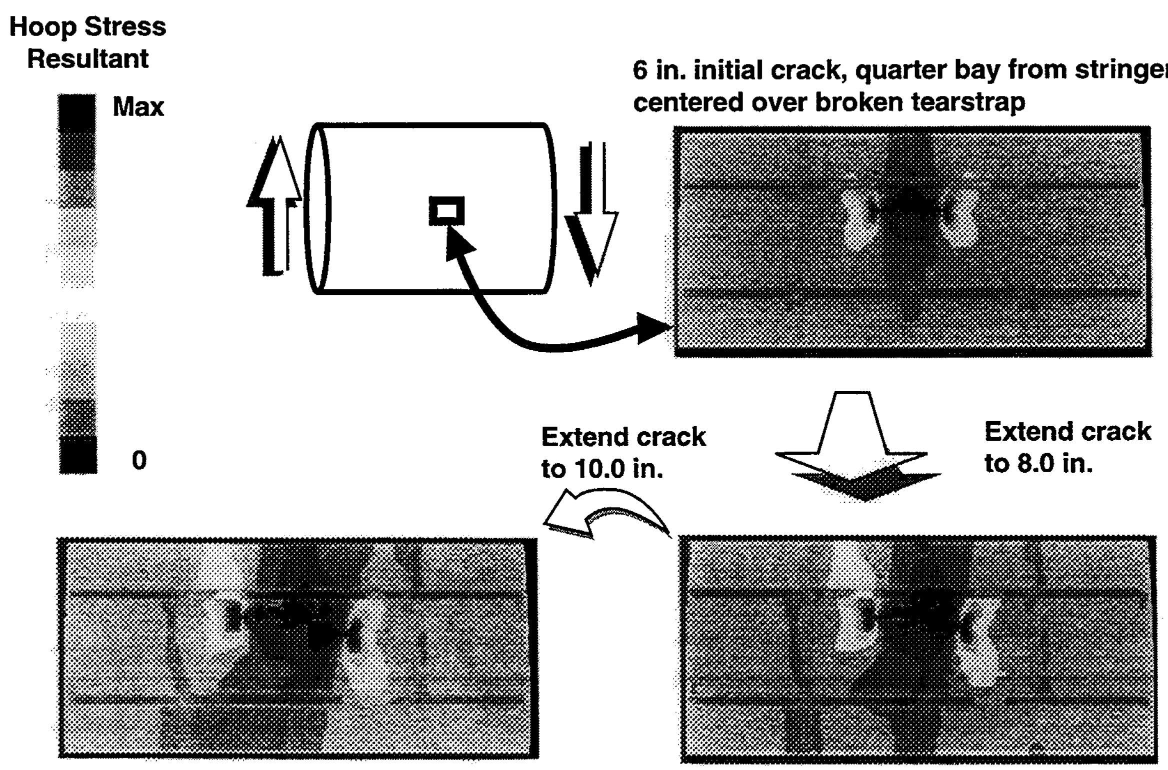

Figure 13. Curvilinear crack extention: internal pressure + shear.

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13. ABSTRACT *(Maximum 200 words)*

NASA has developed a comprehensive analytical methodology for predicting the onset of widespread fatigue damage in fuselage structure. The determination of the number of flights and operational hours of aircraft service life that are related to the onset of widespread fatigue damage includes analyses for crack initiation, fatigue crack growth, and residual strength. Therefore, the computational capability required to predict analytically the onset of widespread fatigue damage must be able to represent a wide range of crack sizes from the material (microscale) level to the global structural-scale level. NASA studies indicate that the fatigue crack behavior in aircraft structure can be represented conveniently by the following three analysis scales: small three-dimensional cracks at the microscale level, through-the-thickness two-dimensional cracks at the local structural level, and long cracks at the global structural level. The computational requirements for each of these three analysis scales are described in this paper.

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