NASA/TM-1998-208975

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December 1998

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National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23681-2199

December 1998

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METHODOLOGY FOR PREDICTING THE ONSET OF WIDESPREAD FATIGUE DAMAGE IN LAP-SPLICE JOINTS

J. C. Newman, Jr., C. E. Harris, R. S. Piascik and D. S. Dawicke
NASA Langley Research Center
Hampton, Virginia USA

ABSTRACT

NASA has conducted an Airframe Structural Integrity Program to develop the methodology to predict the onset of widespread fatigue damage in lap-splice joints of fuselage structures. Several stress analysis codes have been developed or enhanced to analyze the lap-splice-joint configuration. Fatigue lives in lap-splice-joint specimens and fatigue-crack growth in a structural fatigue test article agreed well with calculations from small-crack theory and fatigue-crack-growth analyses with the FASTRAN code. Residual-strength analyses of laboratory specimens and wide stiffened panels were predicted quite well from the critical crack-tip-opening angle (CTOA) fracture criterion and elastic-plastic finite-element analyses (two- or three-dimensional codes and the STAGS shell code).

INTRODUCTION

In the past decade, NASA in collaboration with the Federal Aviation Administration (FAA) and aircraft industry has conducted a program to develop the methodology to predict the onset of widespread fatigue damage (WFD) in lap-splice joints of fuselage structures [1]. The onset of WFD is defined as the life at which fatigue cracking has developed to such an extent that residual strength is reduced below structural requirements. Each aircraft manufacturer has developed in-house durability and damage-tolerance analysis methods that are based on their product development history. To enhance these methods, NASA has adopted the concept of an analytical "tool box" that includes a number of advanced structural analysis codes which represent the comprehensive fracture mechanics capability to predict the onset of WFD. These structural analysis tools have 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 crack-growth properties. Development of these advanced structural analysis codes has been guided by the physical evidence of the fatigue and fracture process in aircraft materials and structures. In addition, critical experiments have been conducted to verify the predictive capability of these codes and to provide the basis for any further methodology refinements.

This paper reviews the advances in fracture-mechanics methodology to predict the onset of WFD in lap-splice joints. To predict the onset of WFD in joints, capability must exist to predict the lives to initiate and grow cracks at rivet-loaded holes and to predict residual strength of joints containing cracks of various lengths at multiple holes. The framework of the NASA program contained elements from all three fracture-mechanics areas: crack initiation (small cracks), fatigue-crack growth (large cracks), and residual strength. For each of these areas, example calculations will be compared to the results of tests to verify the fracture-mechanics criteria and the accuracy of the codes. The crack-initiation (small-crack growth) methodology was verified through fatigue analyses of riveted-lap-splice joints. The fatigue-crack-growth methodology was verified through comparison with crack-growth measurements made on a full-scale structural fatigue test article. Residual-strength methodology was verified on laboratory specimens and wide stiffened panels with multiple-site damage cracking and severe out-of-plane deformations.

STRESS-ANALYSIS CODES

To predict the onset of WFD in lap-splice joints, like those shown in Figure 1, NASA developed or enhanced existing codes to determine the local stresses around the rivet-loaded holes. For two-dimensional (2D) elastic analyses, the FRANC2D [2] and FRANC2D/L [3] codes were enhanced to determine rivet loading and stress-intensity factors for cracked joints. The FADD2D code [4] was developed to determine stress-intensity factors for joints with multiple cracking. The former codes are finite-element based and the latter is a boundary-element code. The codes ZIP2D, ZIP2DL [5] and ZIP3D [6] were used to analyze crack growth under elastic-plastic conditions for 2D and 3D bodies, respectively. For 3D elastic and elastic-plastic analyses, the FRANC3D code [7] and the shell code STAGS [8] were enhanced and linked together to predict both crack trajectories and residual strength of cracked fuselage structures with multiple-site damage.

From detailed studies of the mechanics of crack growth in lap joints, rivet holes act nearly independently of each other for most of the fatigue life of a joint. Stress-intensity factors were obtained for cracks at rivet-loaded holes, as shown in Figure 2, for the following loading conditions: rivet loading (P), remote loading due to rivet loads (S_p), by-pass loading (S_b), remote bending (M) and rivet interference (Δ). Figure 3 shows stress-intensity factors calculated from FRANC2D/L and FADD2D codes [14] for a through-crack at a rivet-loaded hole (see Fig. 2(a)). Rivet and by-pass loading was assumed to be 50% each ($S_p = S_b$) and symmetry boundary conditions were applied to the model edges which simulates one rivet hole in a periodic array of rivet-loaded holes. Normalized stress-intensity factors from the FRANC2D/L and FADD2D codes are shown as symbols as a function of $(c + r)/w_r$, where c is the crack length from the rivet hole, r is rivet hole radius, and w_r is one-half rivet spacing (10 mm). The agreement between the results from the two codes were within 3 percent. An equation (solid curve) was chosen to fit these results. To calculate the growth of a small crack initiating at a critically loaded rivet hole in a lap-splice joint (see Fig. 2), the stress-intensity factors for the various loading conditions were obtained as:

$$K = K_p + K_b + K_M + K_\Delta \quad (1)$$

These stress-intensity factors are used to predict the fatigue lives (using small-crack theory) and crack growth in lap-splice joints in the next section.

FATIGUE-CRACK INITIATION AND SMALL-CRACK GROWTH

Research conducted on small-crack behavior during the last two decades has indicated that "fatigue" of engineering materials is crack propagation from micro-structural defects in the material (see ref. 9). Figure 4 shows a comparison of small- and large-crack data on 2024-T3 aluminum alloy. At high ΔK values the small-crack data [9] and the large-crack (dotted curve) data [10] agreed but small cracks grew at ΔK values much lower than the large-crack threshold (ΔK_{th}). A crack-closure analysis was used to develop the ΔK_{eff}-rate curve (solid lines). This baseline curve was used to predict the growth of small and large cracks using the FASTRAN code [11]. For future use, the crack-closure analyses have been incorporated into the NASGRO life-prediction code [12] which contains a large database on materials and crack configurations.

The fatigue-crack initiation part of the WFD analysis methodology was developed and verified with existing test data on lap-splice joint specimens. A comprehensive test program [13] conducted by the National Aerospace Laboratory (NLR) of The Netherlands determined some of the critical parameters involved in the fatigue of the lap-joint specimens shown in Figure 1(a). Some of the test results (symbols) are shown in Figure 5. These data were for a driven rivet-head diameter of 5 to 5.2 mm. Tests were conducted at a constant mean stress (S_m) and a wide range of alternating stress levels (S_a).

Detailed stress analyses of the riveted lap-joint specimen were conducted [14] using 2D and 3D, elastic and elastic-plastic, finite-element analyses. Fatigue analyses were conducted on the lap-splice-joint specimen shown in Figure 1 using the local stresses, the stress-intensity factors, and small-crack theory. Calculations of fatigue lives of the uniaxially-loaded, flat panels used a fracture-mechanics approach. Stress-intensity factors and crack-opening stresses for small cracks under rivet loading, by-pass loading, and local bending were calculated from some of the codes previously discussed. Effects of hole preparation were accounted for by selection of an "equivalent initial flaw size" (EIFS); and the effects of hole filling by the selection of an "effective" level of interference (Δ) to account for riveting interference, clamp-up, and frictional effects. Plasticity effects were only accounted for in the calculation of crack-opening stresses. Linear elastic stress-intensity factors are calculated even when plastic yielding was present at the rivet hole. The dashed curve shows calculations made with no interference ($\Delta = 0$) using an EIFS of 6 μm (see ref. 9). An effective interference of 5.8 μm was required to fit the mean of the test data (solid curve).

Hartman [13] also conducted variable-amplitude load tests, using a low-high-low load sequence (see insert on Fig. 6) on the same lap joints. The calculated crack-opening stresses for the low-high-low load sequence are shown in Figure 6. A rapid drop in crack-opening stresses occurred after the application of each high stress. The Hartman test data are shown in Figure 7 for the two sets of rivet-head diameters (open and solid symbols). Using the EIFS value, the calculated crack-opening stresses (Fig. 6), and the effective interferences determined from constant-amplitude tests, the solid and dashed lines show the predicted fatigue lives for the larger and smaller rivet-head diameters, respectively. Although there is a large amount of test scatter, the agreement between the mean of the test data and analyses were very good. Based on these comparisons, methods are available to analytically predict the fatigue life of riveted-lap-splice joints using small-crack theory.

FATIGUE-CRACK GROWTH IN LAP-SPLICE JOINT

This section compares the fatigue-crack-growth methodology with test data from a structural fatigue test article. Detailed examinations were conducted on a lap-splice joint removed from a full scale fuselage test article after completing 60,000 pressure cycles [15]. The lap joint had a four-row rivet pattern like that shown in Figure 1(b). The fuselage panel contained a four-bay region that exhibited visible outer skin cracks and regions of crack link-up along the upper rivet row. Destructive examinations revealed undetected fatigue damage in the outer skin, inner skin, and tear strap regions. Outer skin fatigue cracks were found to initiate by fretting damage along the faying surface near the rivet hole. The cracks grew along the faying surface to a length equivalent to two or three times the skin thickness before penetrating the outside surface of the skin. Analysis of fracture surface marker bands produced during the full scale testing revealed that all upper rivet row fatigue cracks contained in a three-bay region grew at similar rates for the same average crack length, as shown by the symbols in Figure 8.

The FASTRAN code was used to predict crack growth in the fuselage lap joint using rivet load, remote stress, and bending stress calculated from finite-element analyses [16]. Examinations of the riveted-lap joint indicated that the interference may be minimal, so the effective interference (Δ) was set equal to zero. Because the material was clad, the initial crack size was selected to be equal to the clad-layer thickness (50 μm). The solid curve shows the calculated rates against the average crack length, $0.5(a + c)$, for a corner crack and c for a through crack. The analysis agreed well with the test results for a corner crack but tended to over predict the rates as the crack penetrated the skin thickness and became a through crack. These results suggests that fracture-mechanics-based methods can be used to predict the growth of outer skin fatigue cracks in lap-splice joint fuselage structures.

RESIDUAL STRENGTH

Now that the crack-initiation (small-crack behavior) and the fatigue-crack-growth methodologies have been developed and verified on laboratory and a full scale structural fuselage test article, the residual-strength methodology will be reviewed. The structural analysis codes under development are being integrated into a methodology for predicting the residual strength of fuselage structure with one or more cracks. The 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 critical crack-tip opening-angle (CTOA) fracture criterion has been experimentally verified to be a valid fracture criterion for mode I stress states in thin and moderately thick (13-mm 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.

In the following sections, the CTOA fracture criterion will be used with various finite-element codes to predict stable tearing and the residual strength of laboratory specimens (restrained from buckling or allowed to buckle) and large-flat-stiffened panels with multiple-site damage cracks that were allowed to buckle.

Laboratory Specimens

The critical crack-tip-opening-angle (CTOA) fracture criterion is a “local” approach to characterizing fracture. An extensive test program [17] has been conducted to experimentally study the characteristics of the CTOA criterion and to establish its validity as a fracture criterion for thin-sheet 2024-T3. Several laboratory-type specimens have been used to measure the CTOA during the fracture process. A high-resolution long-focal-length microscope was used to record the stable-tearing results. The tearing event was then analyzed on a frame-by-frame basis and CTOA was measured. Measurements made on compact C(T) and various size middle-crack tension M(T) specimens are shown in Figure 9. The critical CTOA was relatively insensitive to crack extension after an initial transition region. The initial transition region was caused by 3D effects that occur as the crack tunnels and transitions from flat-to-slant crack growth. Over 50 mm of stable tearing was recorded and the CTOA values were nearly constant (5.8 degs.).

Fracture results from large M(T) specimens (restrained from buckling) are shown in Figure 10 where applied stress is plotted against crack extension. From 3D analysis of smaller M(T) and C(T) specimens, a critical angle of 5.25 degrees was found to fit the test data. The reason(s) for the higher measured angle (see Fig. 9) are still under study. The solid curve is the predicted results from ZIP3D which predict stable tearing and maximum failure stress quite well. Because there is a need to develop 2D codes for faster fracture simulations on the computer, the FRANC2D/L code was enhanced to allow elastic-plastic material behavior and to incorporate the CTOA criterion. Plane-stress analyses (dotted curve) over predict the behavior; and plane-strain analyses (dashed curve) under predict the behavior. Accurate simulations are achieved with the "plane-strain core" concept [18], as shown by the dash-dot curve. The plane-strain core (h_c about equal to the thickness) models the high constraint around a crack tip but allows for the widespread plastic yielding under plane-stress conditions away from the crack tip.

Because the STAGS shell code will ultimately be used to predict the fracture behavior of cracked fuselage structures, the code needed to be verified on laboratory specimens that were restrained from buckling or allow to buckle. Buckling of an M(T) specimen, with severe out-of-plane deformations, is similar to the bulging of a cracked fuselage under pressure. The STAGS code and the CTOA criterion were used to predict the effects of buckling on the residual strength of aluminum alloys and steel specimens [19]. A comparison of these results are shown in Figure 11. The failure load under buckling (P_b) normalized by the failure load with no buckling (P_{nob}) are plotted against the crack-length-to-thickness ratio (c/B). The curves show the predicted load ratios as a function of c/B for the two materials. The results agreed quite well, even though the STAGS code did not have the "plane-strain core" option. Later, the plane-strain core option was incorporated into STAGS and the enhanced code will be used in the next section to analyze flat-stiffened panels with single cracks and multiple-site damage (MSD) cracking.

Flat Stiffened Panels

NASA and the FAA jointly designed and conducted fracture tests on 1016-mm wide sheets made of 1.6-mm thick 2024-T3 aluminum alloy with and without stiffeners [20]. Some of the specimens had five 7075-T6 aluminum alloy stiffeners (2.2-mm thick) riveted on each side of the sheet, as shown in Figure 12(a). The central stiffeners were cut along the crack line. Open holes were machined into the sheet at the required rivet spacing along the crack line but rivets were not installed. Five different crack configurations were tested: a single center crack, a single center crack with an array of 12 holes on either side of the lead crack, and a single center crack with three different equal MSD cracking (0.25, 0.76 and 1.3-mm) at the edge of each hole, see Figure 12(b). For each crack configuration, identical specimens were tested with and without riveted stringers. All tests were conducted under stroke control. Measurements were made of load against crack extension.

Comparisons of measured and predicted load against crack extension for a stiffened panel test with a single crack and a test with a single crack and MSD are shown in Figures 13 and 14, respectively. The CTOA (5.4 deg.) was determined from laboratory specimens restrained from buckling [21]. The stiffened panels were allowed to buckle. The STAGS analyses with the plane-strain core (h_c = 2 mm) compared extremely well with the test data (symbols). These results demonstrate that the residual-strength analysis method can predict stable crack growth and failure loads for complex structure.

CONCLUDING REMARKS

A comprehensive analytical methodology has been developed for predicting the onset of widespread fatigue damage (WFD) in lap-splice joints and complex structure. The determination of life (cycles) related to the onset of WFD includes analyses for crack initiation, fatigue-crack growth, and residual strength. Each area was validated with tests and demonstrated the capabilities of the analysis tools. These tools, taken together, provide the methodology to predict WFD in structures.

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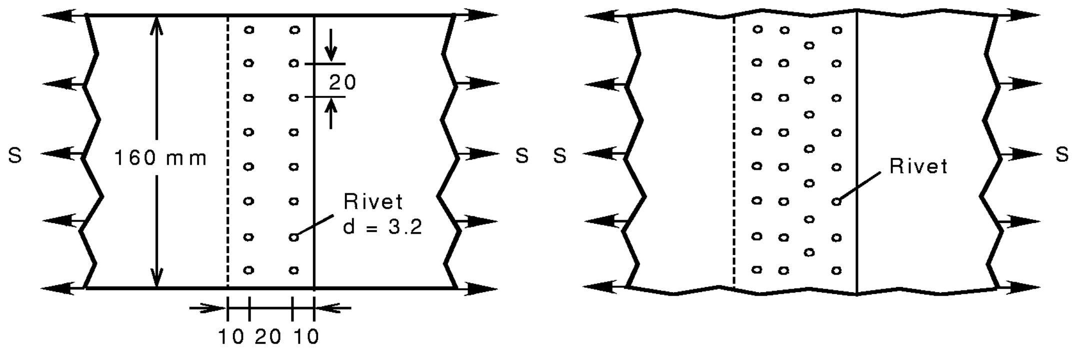


(a) NLR lap-joint specimen [13] (b) Lap joint in fatigue test article [15]

Figure 1 Laboratory and structural fatigue test article lap-joints analyzed

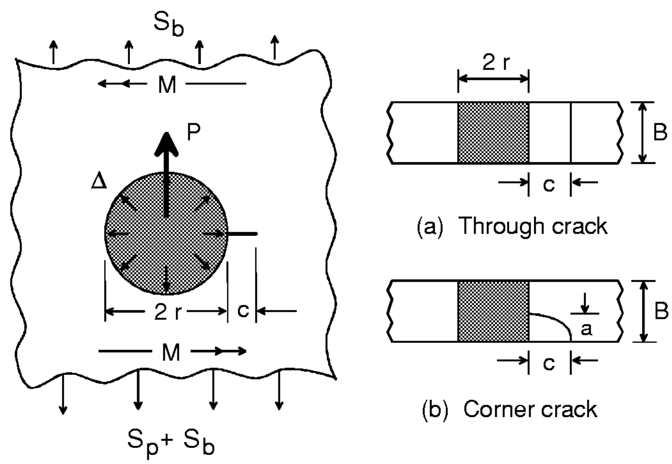


Figure 2 Crack configurations and loadings in fracture-mechanics analyses

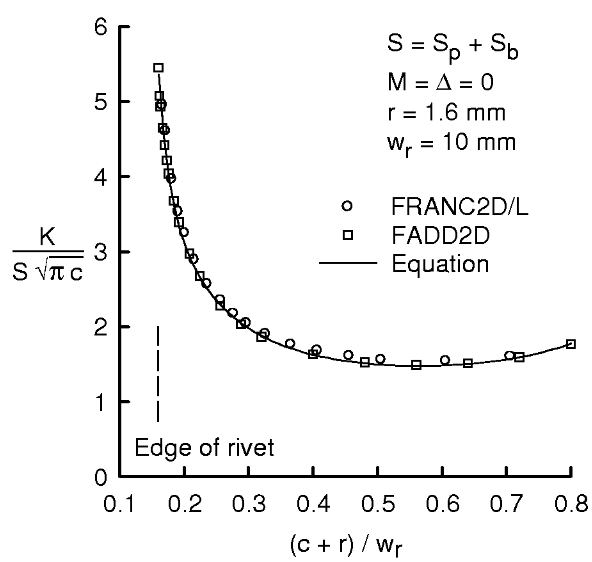


Figure 3 Stress-intensity factors for through crack at rivet-loaded hole [14]

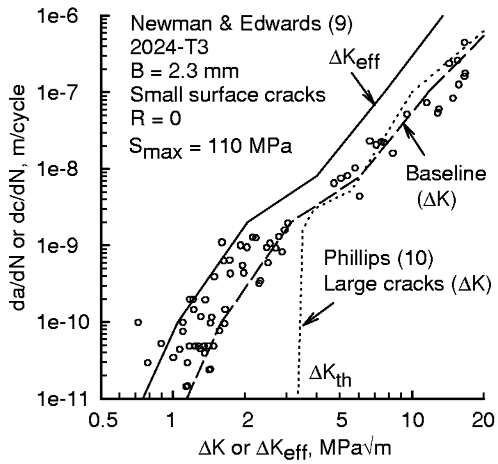


Figure 4 Small- and large-crack growth rate data on 2024-T3 alloy

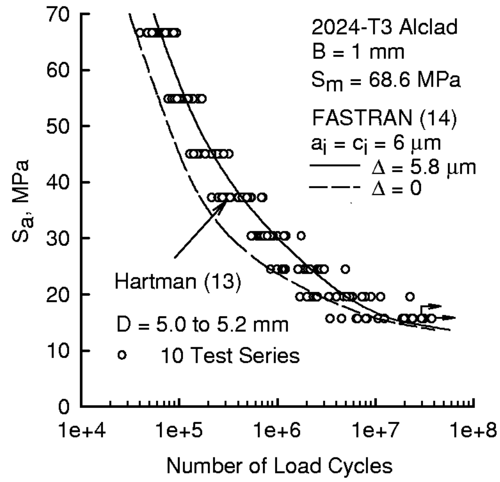


Figure 5 Measured and calculated fatigue lives for lap-joint specimen

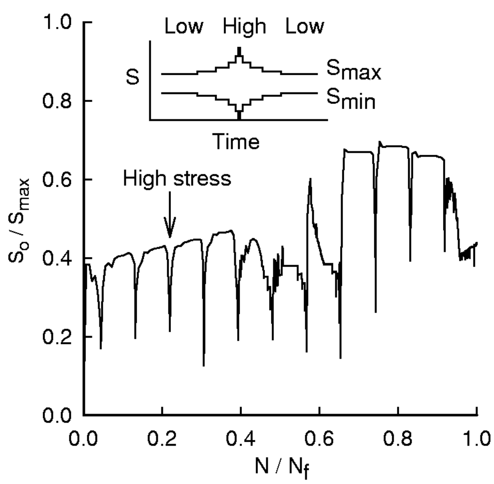


Figure 6 Crack-opening stresses under low-high-low sequence [14]

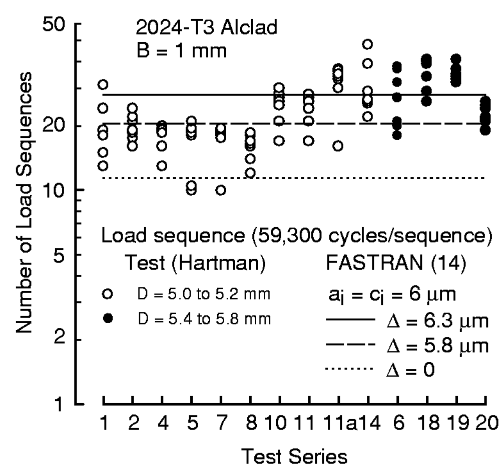


Figure 7 Measured [13] and predicted fatigue lives for lap-joint specimen under low-high-low sequence

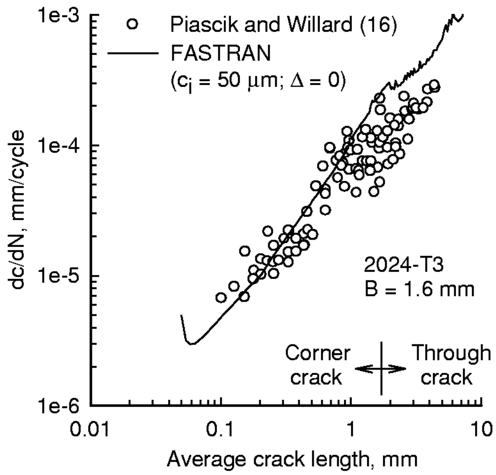


Figure 8 Measured and predicted crack-growth rate of structural fatigue test article

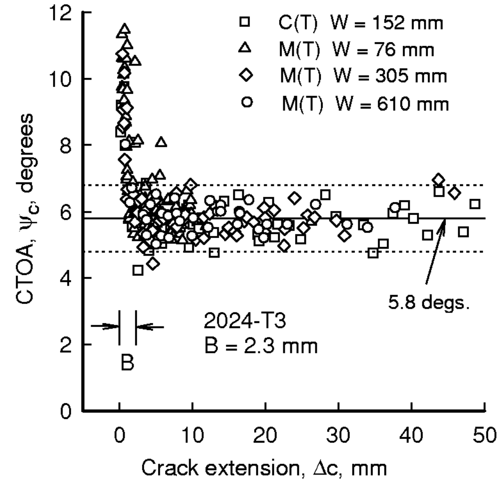


Figure 9 Measured critical crack-tip opening angle on 2024-T3 alloy

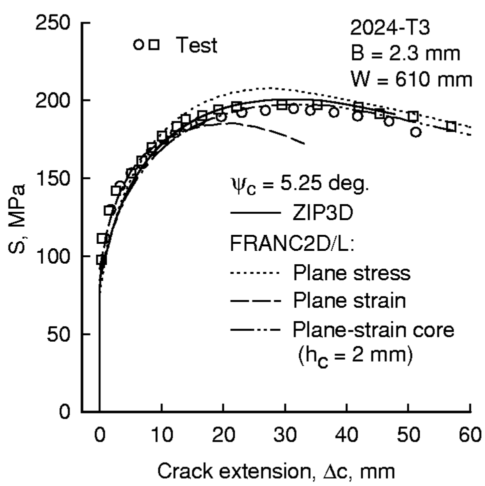


Figure 10 Measured and predicted applied stress against crack extension for 2024-T3 alloy

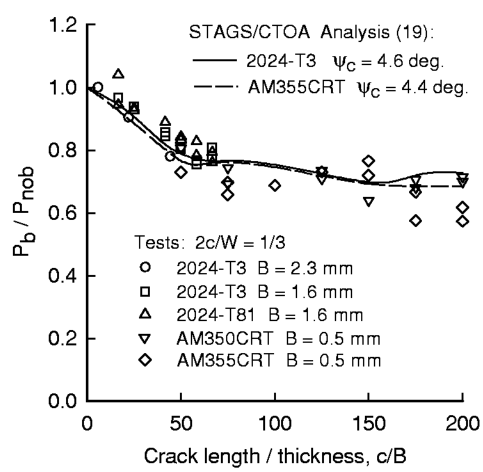
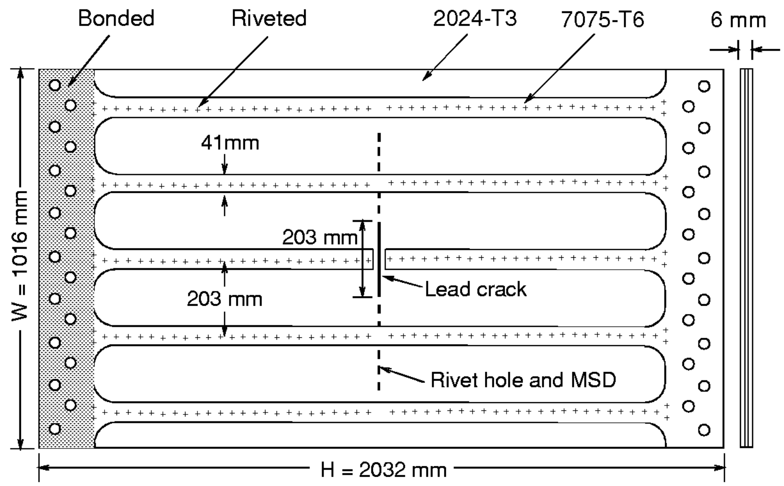
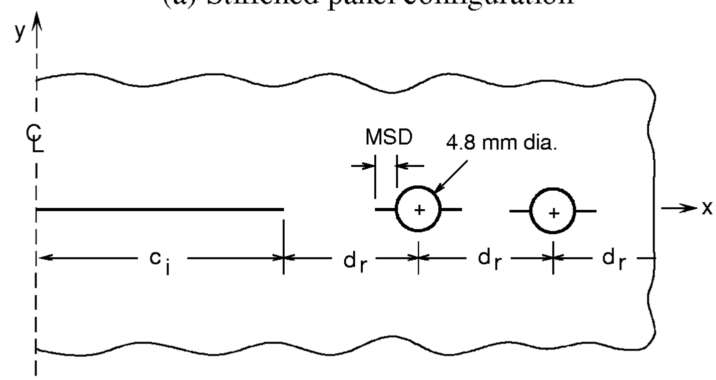


Figure 11 Measured and predicted effects of buckling on residual strength for aluminum alloys and steel



(a) Stiffened panel configuration



(b) Typical open-hole and multiple-site damage cracks

Figure 12 FAA/NASA stiffened panel and typical multiple-site damage cracks [20]

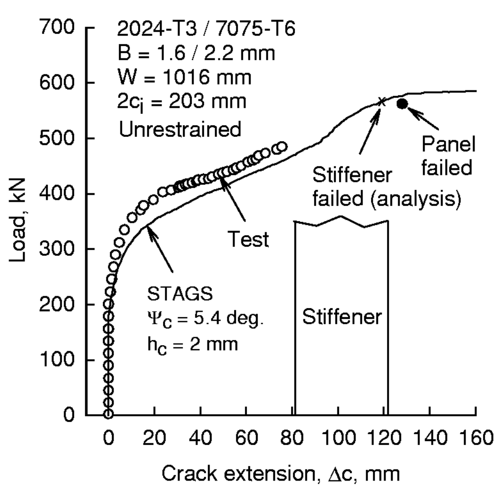


Figure 13 Measured and predicted failure of stiffened panel [21]

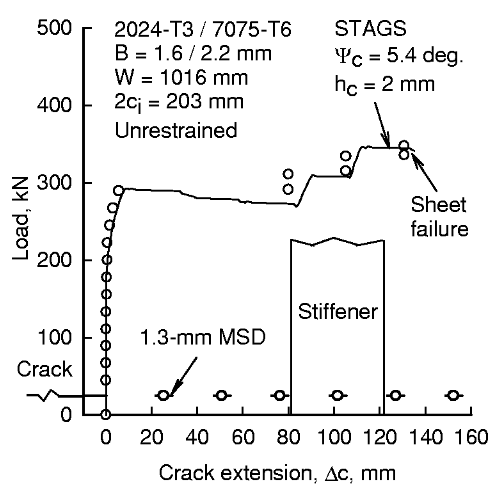


Figure 14 Measured and predicted failure of panel with MSD cracks [21]

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Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	December 1998	Technical Memorandum

4. TITLE AND SUBTITLE
Methodology for Predicting the Onset of Widespread Fatigue Damage in Lap-Splice Joints

5. FUNDING NUMBERS
WU 522-18-11-01

6. AUTHOR(S)
J. C. Newman, Jr., C. E. Harris, R. S. Piascik, and D. S. Dawicke

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
NASA Langley Research Center
Hampton, VA 23681-2199

8. PERFORMING ORGANIZATION REPORT NUMBER
L-17799

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA/TM-1998-208975

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified-Unlimited
Subject Category 24 Distribution: Standard
Availability: NASA CASI (301) 621-0390

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

NASA has conducted an Airframe Structural Integrity Program to develop the methodology to predict the onset of widespread fatigue damage to lap-splice joints of fuselage structures. Several stress analysis codes have been developed or enhanced to analyze the lap-splice-joint configuration. Fatigue lives in lap-splice-joint specimens and fatigue-crack growth in a structural fatigue test article agreed well with calculations from small-crack theory and fatigue-crack growth analyses with the FASTRAN code. Residual-strength analyses of laboratory specimens and wide stiffened panels were predicted quite well from the critical crack-tip-opening angle (CTOA) fracture criterion and elastic-plastic finite-element analyses (two- or three-dimensional codes and the STAGS shell code).

14. SUBJECT TERMS
Cracks; Fatigue; Fracture; Fracture mechanics; Aluminum; Lap-splice joints; Stress-integrity factor

15. NUMBER OF PAGES
20

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	UL