EXPERIMENTAL VERIFICATION OF THE ANALYTICAL METHODOLOGY TO PREDICT THE RESIDUAL STRENGTH OF METALLIC SHELL STRUCTURE

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ABSTRACT

Experimental and analysis results for a curved, stiffened aluminum fuselage panel tested in a combined loads test machine with combined internal pressure, axial compression, and torsional shear loads are described. The experimental and analytical strain results for the panel with and without discrete source damage are presented. The effect of notch tip geometry on crack growth predictions is addressed. The crack growth trajectory predictions for the panel are presented for the applied loading conditions at failure.

1 INTRODUCTION

Understanding the response of aircraft structures subjected to combined loading conditions representative of actual flight conditions is an important aspect of designing aircraft structures. Since testing full-scale fuselage structures is expensive, curved panels representative of these structures are often tested to study the response of full-scale structures. One approach for testing curved panels is to attach a panel to a cylindrical shell fixture with the radius of the panel and with a cutout to accommodate the test panel. The panel is then subjected to the intended combined loading conditions by loading the cylindrical shell fixture. The correct boundary conditions can be imposed on the test specimen more readily using this approach. In spite of this advantage, this approach is not the preferred approach if the requirement is to test panels of different radii than the cylindrical shell fixture, which would require cylindrical shell fixtures of different radii. An inexpensive test fixture that can be used to test curved stiffened panels of different radii and frame spacings would be useful for testing such panels. Such a test fixture could also be used to study the response of damaged structures, and to validate analysis methods that predict the residual strength of structures subjected to combined loading conditions.

The objectives of the present paper are to present experimental and analysis results for a metallic fuselage panel tested in a combined loads test machine, and to demonstrate analytical methods used for predicting damage initiation and growth in metallic structures with long notches and subjected to combined loads.

2 TEST SETUP

The COmbined Loads Test Systems (COLTS) facility at NASA Langley Research Center includes a combined loads test machine which can be used for testing curved panels, fuselage barrels, and wing sections subjected to combined mechanical, pressure and thermal loading conditions. The combined loads test machine configuration with a cylindrical shell mounted between the test machine platens is illustrated in Fig. 1. This facility has been designed to accommodate cylindrical shells with a maximum diameter of 15 feet and a maximum length of 45 feet. A10-foot-long D-box test fixture, as shown in the inset of Fig. 1, can be located between the platens to test curved stiffened panels subjected to combined compression or tension, torsional, shear, and internal pressure loading conditions. The test machine is equipped with 10 hydraulic actuators that can apply combined mechanical loads of up to 2,700 kips of axial load, 600 kips of shear load, and 2,900 ft-kips of torsion to the test specimen.

The main issue associated with testing a curved panel in this D-box fixture is that, as the panel is loaded, the fixture must provide the appropriate stress distributions in the panel to represent those in a complete shell subjected to the same loading conditions. This requirement must be satisfied for both undamaged and intentionally damaged panels for the panel tests to be meaningful. The design requirements and features for the D-box test fixture that ensure proper test conditions are presented in Ref. 1.

3 TEST SPECIMEN

The aluminum test panel evaluated in the present paper is shown in Fig. 2. The panel is approximately 10-ft long and 10-ft wide. Its 0.084-in.-thick skin is made from 7475-T61 aluminum alloy. The panel has a frame spacing of 20 inches and a stringer spacing of 8.5 inches. Single and rosette strain gages were used to monitor the panel axial, hoop, and shear strains. Strip gages were also mounted on the panel to record strain data at the tips of a notch in the panel as crack growth initiated and propagated. A photograph of the test panel attached to the D-box test fixture in the COLTS test machine is shown in Fig. 3.

4 TEST CONDITIONS

The undamaged panel was subjected to combinations of internal pressure, axial tension, axial compression, and torsional shear loads. After completing the undamaged panel tests, a 20-inch-long longitudinal notch was cut into the panel with a saw blade. The notch extended from the center of one

skin bay to the next skin bay and severed the frame in between. The damaged panel was tested to failure by sequentially applying a combination of internal pressure, axial compression, and torsional shear loads to the panel.

5 RESULTS AND DISCUSSION

The undamaged and damaged test panels were analyzed using the Structural Analysis for General Shells (STAGS) code.[2] The finite element model of the damaged panel is shown in Fig. 4. Experimental and analysis results for the panel buckling response, and axial, hoop, and shear strain distributions for the damaged and undamaged test panels are summarized in Ref. 3. The analytical axial strain results at Location A (Fig. 4) for the panel are compared with experimental results in Fig. 5 for a loading condition with 3.9 psi of internal pressure, 149 kips of axial compression, and 2,500 in-kips of torsion.

Crack growth initiated in the test for a loading condition with 3.9 psi of internal pressure, 149 kips of axial compression, and 5,200 in-kips of torsion. The crack growth trajectory is illustrated in Fig. 6. Yielding was recorded near the crack tips at lower load levels, but there was no visible crack growth observed until the crack extended from each end of the initial sawcut notch by dynamic fast fracture. The observed crack growth trajectory was not collinear with the panel longitudinal axis, but was inclined at clockwise angles of 51° and 47° at the two ends of the notch, respectively. This non-self-similar crack growth trajectory extended approximate 3.5 inches in length from each crack tip.

A finite element model representing the initially straight notch was created to determine analytically the direction of crack extension, and a geometrically nonlinear analysis was conducted for the loads corresponding to the observed crack extension in the test. A contour plot of the hoop stress resultants corresponding to the combined loading condition just prior to crack initiation at the notch is shown in Fig. 7. The torsional load applied to the panel causes asymmetry in the panel deformations and stresses. The linear elastic fracture mechanics stress-intensity factors are computed from strain-energy-release rates for the maximum applied load by applying the modified crack closure integral method (e.g., Refs. 4-5). The crack turning angle is then computed from the stress-intensity factors using the maximum tangential stress criterion defined by Erdogan and Sih (Ref. 6). These analytical methods predicted crack turning angles for the two ends of the notch to be 47° and 46°, respectively.

Elastic-plastic stable-tearing analyses were conducted to predict the crack extension behavior. The CTOA crack tip opening angle criterion (e.g., Refs. 7-8) was used in the analysis with a plane strain core to simulate three-dimensional constraint effects with two-dimensional elements. The CTOA criterion is based on the concept that the crack tip will extend when the crack tip opening angle, computed a fixed distance behind the crack tip, exceeds a constant material and thickness dependent allowable value. The nonlinear stress-strain curve for the 0.084-in.-thick panel skin material was obtained from Ref. 9. The skin fracture parameters, $CTOA_{cr}$ = 4.4° and a plane strain core height h_c = 0.08 in., were selected based on elastic-plastic residual strength analyses that simulated the response of a middle-crack tension test reported in Ref. 10. A previous study[8] described a case where crack growth from a sawcut notch similar to the one in the test panel required an opening angle 3.26 times larger for initiation than for subsequent crack extension. Thus, $CTOA_{cr}$ was estimated to equal 14.35° for crack initiation, and 4.4° for subsequent crack extension for the residual strength analysis of the test panel. The finite element model was modified to introduce an inclined crack trajectory emanating from each end of the initial notch. The trajectory angles were predicted from the elastic analysis, and the mesh was refined to provide 0.04-inch-long elements along the crack trajectory.

Residual strength analyses were conducted for the test loading conditions. The internal pressure and axial compression loads were applied first and held constant, and then the torsional load was added until the crack extended. A typical result from the nonlinear residual-strength stable-tearing analysis for this model is shown in Fig. 8. This figure shows the plastic hoop strains superimposed on the deformed panel shape for a load corresponding to a crack extension of 0.40 inches at one crack tip. The analysis predicted nearly identical crack extension behavior for the crack tips at each end of the notch. The load versus crack extension results from the test and analysis are compared in Fig. 9. The abscissa represents the crack extension at one end of the notch, and the ordinate represents the applied torsional load. The test data are shown in the plot by the solid symbols, which suggest fast fracture at a specific load value. The load versus crack extension behavior obtained from the analysis with $CTOA_{cr}$ = 14.35° for crack initiation, and $CTOA_{cr}$ = 4.4° for subsequent extension is shown by the solid line in the figure. The crack growth behavior predicted by an analysis with $CTOA_{cr}$ = 4.4° for crack growth initiation is illustrated by the dashed line in Fig. 9. Both analyses predict an increase in the crack growth rate as the applied torsional load approaches the test failure load. The analysis that includes the higher value for the CTOA to account for the initial sawcut notch has a crack initiation behavior that more closely represents the experiment. To avoid the influence of the notch tip geometry on crack initiation, an initial sharp crack should be developed by subjecting the specimen to fatigue loads prior to conducting the residual strength test.

6 CONCLUDING REMARKS

An experimental and analytical study has been conducted to understand the response of a curved, stiffened aluminum aircraft fuselage panel subjected to combined internal pressure, axial compression, and torsional loads. The good correlation between the analytical and experimental strain results suggests that the panel nonlinear response was accurately predicted by the analysis. The results suggest that that an accurate representation of the notch tip geometry is important for accurately predicting the crack initiation and residual strength of the panel. The crack trajectory prediction that corresponds to the panel test

failure load provided a good estimate for the crack turning angle.

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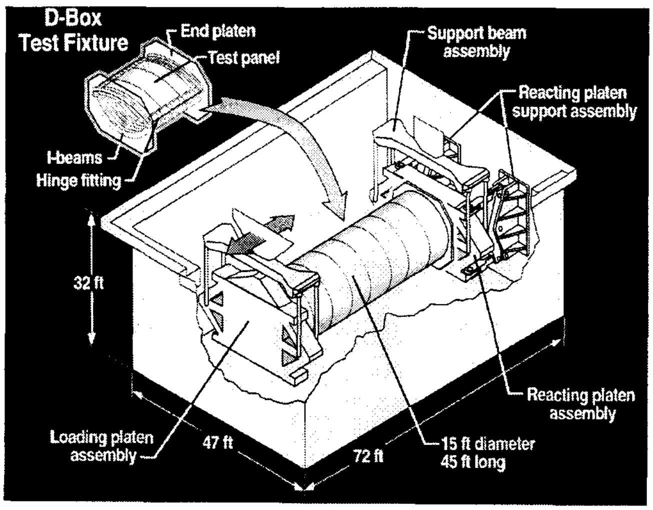

Figure 1. Schematic of the combined loads test machine.

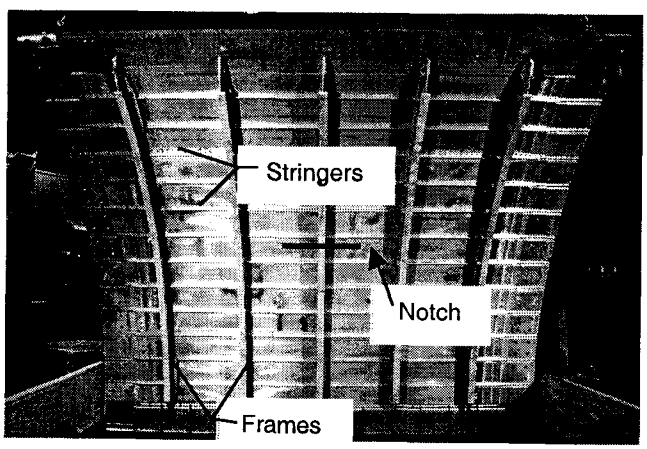

Figure 2. Stiffened aluminum fuselage test panel.

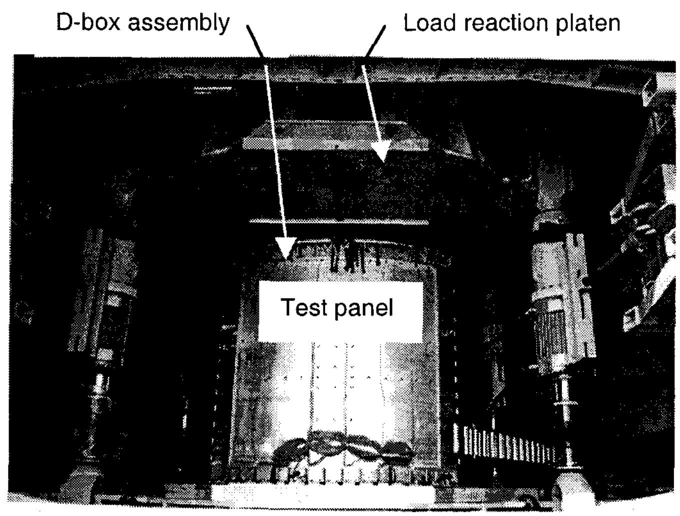

Figure 3. D-box test fixture assembly located in the combined loads test machine.

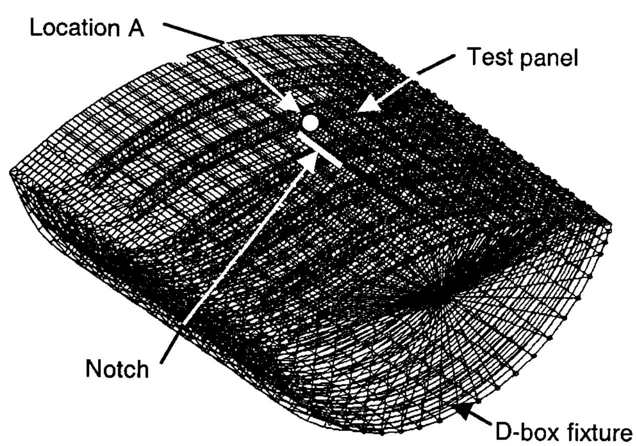

Figure 4. Finite element model of the test panel and the D-box test fixture.

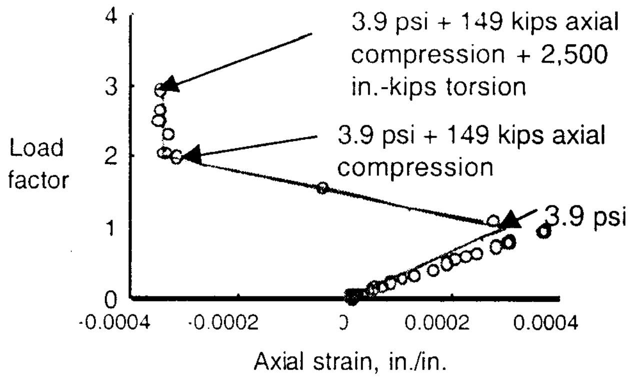

Figure 5. Axial strain at Location A (Fig. 5) for sequential application of internal pressure, axial compression and torsional loading conditions.

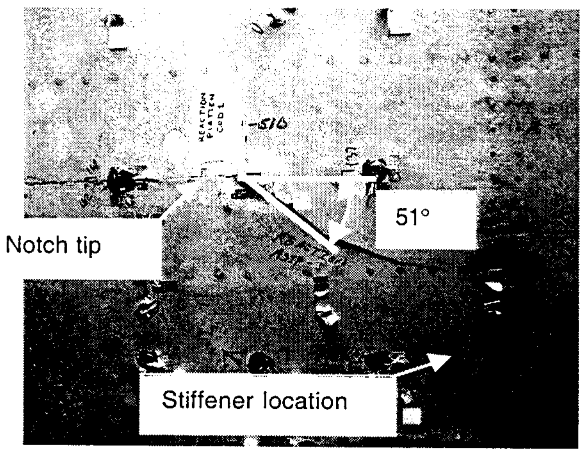

Figure 6. Crack growth trajectory for the test panel with a combined loading condition of 3.9 psi of internal pressure, 149 kips of axial compression, and 5,200 in-kips of torsional loads.

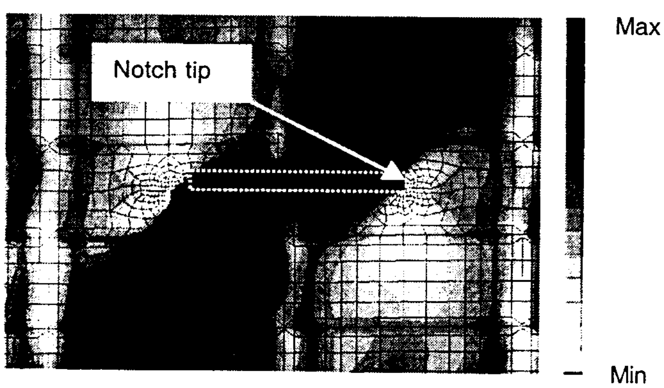

Figure 7. Analytical hoop stress resultant contours around the notch tip for the combined loading condition just prior to crack initiation.

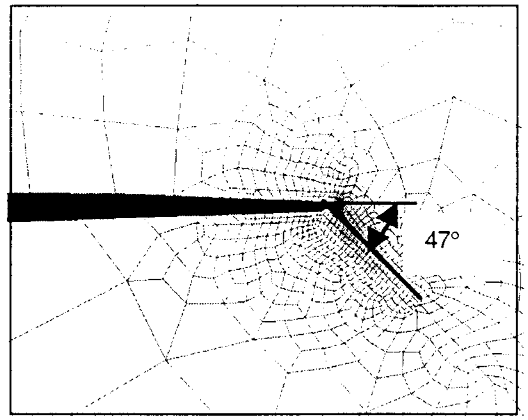

Figure 8. Analytical plastic hoop strain results for the failure loading condition and corresponding to a crack extension of 0.4 inches.

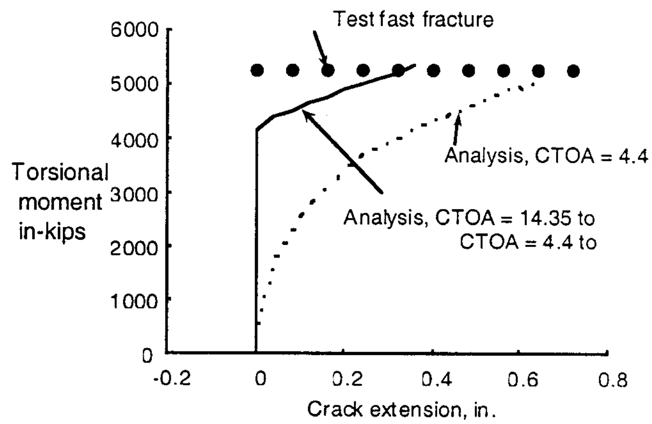

Figure 9. Analytical crack extension results for an increasing amount of torsional load on the panel preloaded with 3.9 psi of internal pressure and 149 kips of axial compression loads.