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DESIGN, ANALYSIS, AND FABRICATION OF A PRESSURE BOX TEST FIXTURE FOR TENSION DAMAGE TOLERANCE TESTING OF CURVED FUSELAGE PANELS[1]

P. J. Smith, J. B. Bodine, C. H. Preuss
Boeing Commercial Airplane Group.

W. J. Koch
Boeing Computer Services

516-24

ABSTRACT

A pressure box test fixture was designed and fabricated to evaluate the effects of internal pressure, biaxial tension loads, curvature, and damage on the fracture response of composite fuselage structure. Previous work in composite fuselage tension damage tolerance, performed during NASA contract NAS1-17740, evaluated the above effects on unstiffened panels only. This work extends the tension damage tolerance testing to curved stiffened fuselage crown structure that contains longitudinal stringers and circumferential frame elements. The pressure box fixture was designed to apply internal pressure up to 20 psi, and axial tension loads up to 5000 lb/in, either separately or simultaneously. A NASTRAN finite element model of the pressure box fixture and composite stiffened panel was used to help design the test fixture, and was compared to a finite element model of a full composite stiffened fuselage shell. This was done to ensure that the test panel was loaded in a similar way to a panel in the full fuselage shell, and that the fixture and its attachment plates did not adversely affect the panel.

INTRODUCTION

The objective of Boeing's Advanced Technology Composite Aircraft Structures (ATCAS) program (NAS1-18889) is to develop an integrated technology and demonstrate a confidence level that permits the cost- and weight-effective use of advanced composite materials in primary structures of future commercial transport aircraft. The emphasis of the program is on pressurized fuselages. A significant portion of a typical commercial transport fuselage is designed by either tension from internal pressure and/or flight loads (see Figure 1), therefore the specific emphasis of this paper is on this tension critical structure such as the fuselage crown. The approach of the ATCAS program was to build on tension fracture coupon data with larger unstiffened and stiffened panel analyses and tests to culminate with the analysis and test verification of configured crown panels.

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1 This wok was funded by Contract NAS1-18889, under the direction of J. G. Davis and W. T. Freeman of NASA Langley Research Center

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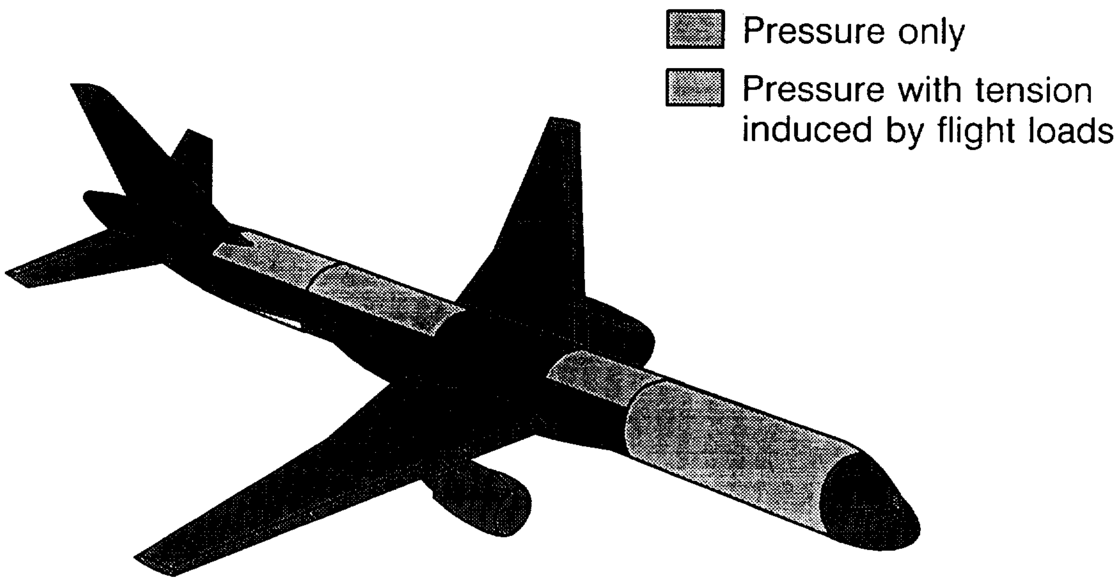

Figure 1. Tension-dominated commercial fuselage structure.

The effects of internal pressure on the tension damage tolerance and pressure containment of composite structure are not understood, and it is essential that this insight be gained if composite materials are to be used for the basic fuselage monocoque of commercial transport aircraft.

The design envelope for the ATCAS development program is that fuselage section of a wide body aircraft immediately aft of the main gear wheel well, called section 46 on Boeing airplanes. This section 46 is 32 feet long and 122 inches radius. After initial studies the section was divided into three quadrants, crown, side and keel, these being shown in Figure 2. As was seen in Figure 1 the crown quadrant is designed by hoop tension due to internal pressure and axial tension due to flight loads at the forward end, and hoop tension in the aft part of section 46. The fuselage aft of the wing is loaded by the wing and horizontal stabilizer as a beam in bending during flight maneuvers with typically the crown in tension and the keel in compression, with load reversal during negative flight maneuvers.

The presence of the cutouts in the fuselage for the wing center section and the main landing gear wheel well just ahead of the section 46 produce high axial loads from flight maneuvers at the forward end which decay toward the rear of the section. These flight induced loads can be present with or without the internal cabin pressure and the structure needs to be evaluated for those load combinations which may be critical. The flight induced axial loads are augmented by the bulkhead loads from the cabin pressure, and negated slightly by the Poisson's effect of the hoop loads induced by the cabin pressure. For a typical metal commercial fuselage the forward end of the section 46 crown tends to be critical for axial flight loads combined with internal pressure, and the aft end is usually critical for the hoop tension generated by the internal cabin pressure alone. Fuselage structure constructed with composite materials, which have different failure modes than metal, may be critical for other combinations of flight and pressure loads, such as axial tension and pressure at the forward end of the section 46, and axial compression with pressure at the rear of the section.

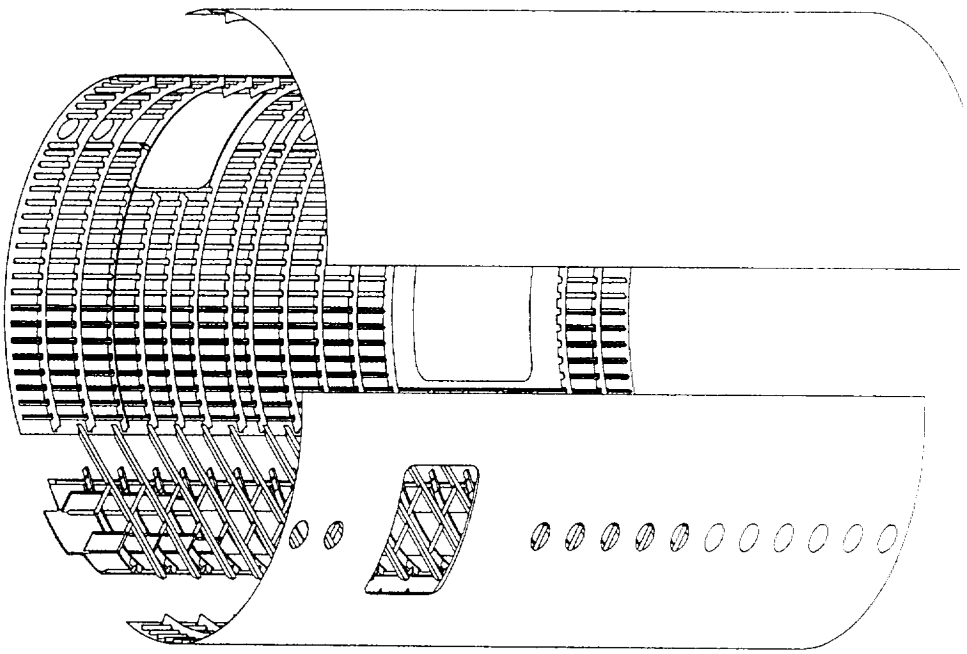

Figure 2. Quadrant approach to Boeing fuselage section 46.

Figure 3 summaries the ultimate strength and damage tolerance requirements for commercial aircraft primary structures. These requirements are based on those of the Federal Aviation Requirements, specifically those of Part 25 Airworthiness Standards for Transport Category Airplanes, paragraph 25.301 through 25.571. The requirements for damage tolerance and pressure containment are set down in paragraph 25.571 which states that an evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage will be avoided throughout the operational life of the airplane. These damage tolerance strength requirements are commonly referred to as fail-safe and include residual strength requirements for discrete source damage sustained by the airframe that the crew are aware of and for which the flight loads are reduced. However, fuselage pressure cabin structure must be able to withstand discrete source damage, such as that inflicted by uncontained engine fragments, with normal operating cabin pressure. Consequently the likely damages from discrete sources may be significantly larger than damages that may go undetected until normal maintenance inspections, thus making fuselage pressure designed structure more critical than other primary structural components. To this end testing of curved fuselage configured panels under axial and internal pressure induced loads is essential in order to build the confidence level necessary to allow composite materials to be effectively utilized in commercial transport fuselages.

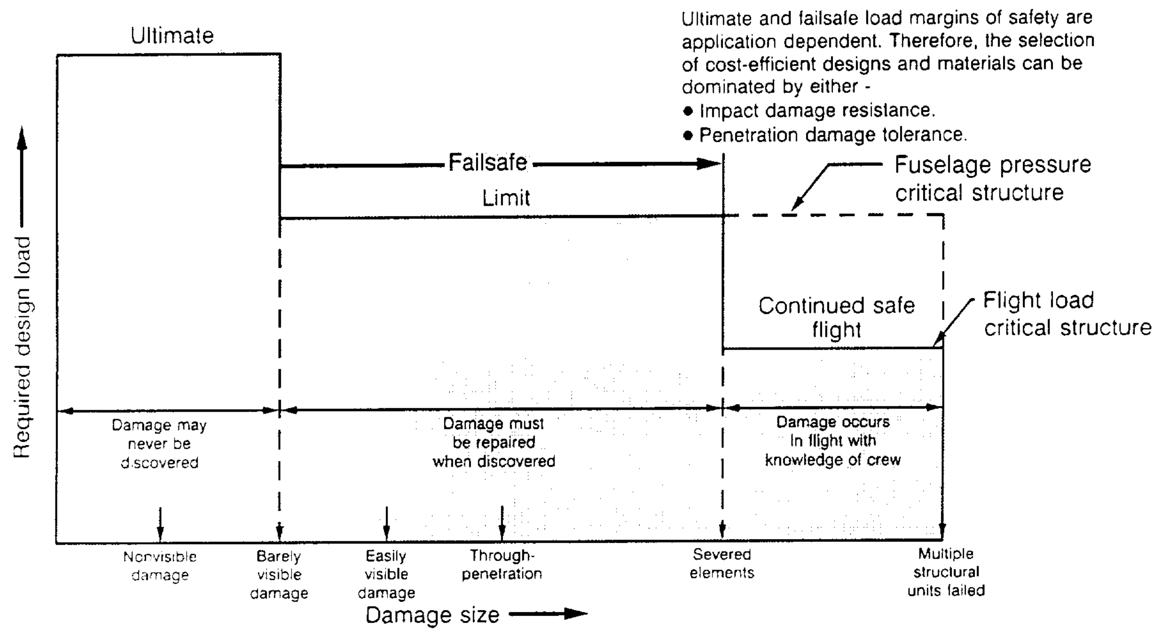

Figure 3. Strength and damage tolerance requirements for commercial aircraft primary structure.

TEST FIXTURE REQUIREMENTS

The maximum design loads considered for the pressure box fixture were derived from the initial design studies of the crown area of an aft section of a 244 inch diameter commercial transport fuselage. These design loads are summarized in Table 1 and show both ultimate and fail-safe loads. The maximum ultimate axial load was taken from the forward end of the section at the top centerline. This axial load, from an ultimate vertical gust load case with internal cabin pressure, includes the bulkhead load of PR/2 and the negating Poisson's effect of the cabin pressure. The pressure associated with this ultimate axial load is derived from the maximum cabin pressure relief valve setting combined with the expected external aerodynamic pressure. The maximum internal pressure of 18.2 psi is obtained from the maximum pressure relief valve setting alone multiplied by a factor of 2.0. The ultimate loads are those that may be applied to panels that have no damage that is visible to the naked eye. This includes impact damage up to barely visible, the upper level being 200 foot-pounds of energy inflicted by a 1.0 inch diameter steel ball.

The maximum fail-safe axial load is simply that load from the ultimate gust case above with the 1.5 factor removed. The pressure combined with this maximum fail-safe load is the normal operating cabin pressure plus the expected external aerodynamic pressure. The maximum fail-safe load case for pressure acting alone is the normal operating cabin pressure plus the expected external aerodynamic pressure both multiplied by a 1.15 factor.

Load condition	Axial load, lb/in	Hoop load, lb/in
Ultimate [1]		
a. 2 x (maximum pressure relief valve setting) = 18.2 psi	---	2220
b. 1.5 x (maximum flight loads + (maximum pressure relief valve setting + expected external aerodynamic pressure = 9.15 psi))	5000	1675
Failsafe [2]		
a. limit flight loads + (normal operating differential pressure + expected external aerodynamic pressure = 8.75 psi)	3333	1067
b. 1.15 x (normal operating differential pressure + expected external aerodynamic pressure) = 10.1 psi	---	1228

[1] Maximum test-box loads for undamaged test panels or panels with barely visible damage.

[2] For visibly damaged structure (i.e., skin, frames, and stringers severed).

Table 1. Pressure box fixture design loads.

The pressure box fixture was designed to be capable of applying the above loads multiplied by a safety factor of 2.0. The fixture was strength checked for positive margins of safety with this factor applied against the yield strengths of the materials used. This conservative procedure ensures that not only does the fixture have more than adequate strength, but fixture deformations under load will be minimized. The 2.0 factor also ensures adequate durability under repeated static load cycles.

TEST PANEL CONFIGURATIONS

The pressure box fixture will be used to test different types of curved panels. The first panel that has actually been tested in the fixture is a 122 inch radius curved panel, 63 inches arc width by 72 inches long. The panel, shown in Figure 4, was tow-placed with AS4[2]/938[3] material and is stiffened by three cobonded circumferential tear-straps fabricated from AS4/3501-6[4] fabric. These tear-straps simulate the skin flanges of circumferential cobonded frames. The intent of this panel configuration, which would not be suitable for an actual fuselage shell which has to be stiffened by frames and stringers or sandwich core, is to provide some insight into the effects of damage growth in skins in the presence of frame flanges. When considering typical metal stiffened fuselage shell

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2 AS4 is a graphite fiber system produced by Hercules, Inc.

3 938 is a resin system produced by ICI/Fiberite.

4 3501-6 is a resin system produced by Hercules, Inc.

structures, the presence of the stringers and frames and their method of attachment to the skins have significant effects on damage growth and damage tolerance of the total skin panels. These effects are labeled configuration factors and were derived from elastic-plastic analysis or tests to aid in calculating the residual strength of the structure in the presence of differing amounts of damage. The panel with the tear-straps, representing frame flanges, is intended to provide a link between the unstiffened panel fracture strength analysis and tests, and the stiffened panels.

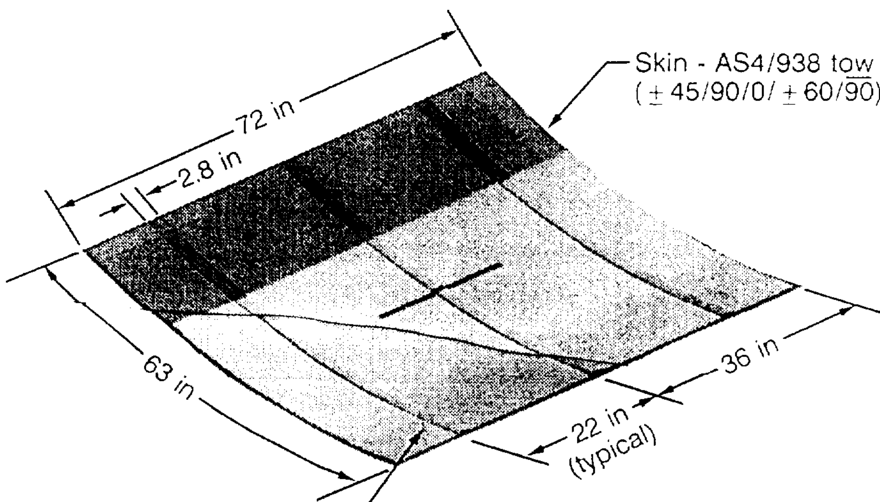

Figure 4. Curved tear-strap test panel configuration.

The second test panel configuration is representative of the crown structure that has been optimized for weight and cost in the ATCAS program. The skin, fabricated of tow-placed AS4/938 material, is stiffened by four longitudinal cocured enclosed hat-stringers of the same material, and three triaxially braided resin transfer molded circumferential frames fabricated from AS4/1895[5]. Figure 5 presents this panel configuration with the frames mechanically attached to the skins. The second and third panels will have four longitudinal stringers and three frames cobonded to the skins. The stiffened panels will provide insight on configuration factors for stiffened panels, loaded with internal pressure and/or axial loads, that have through penetration type damages. These damages will include skin, skin and frame, and skin and stringer severed on different panels. The differences between the cobonded and mechanically fastened frame flange/skin interfaces will provide data on their configuration effects on the damage tolerance of the panels.

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[5] 1895 is a resin system produced by Shell Development Co.

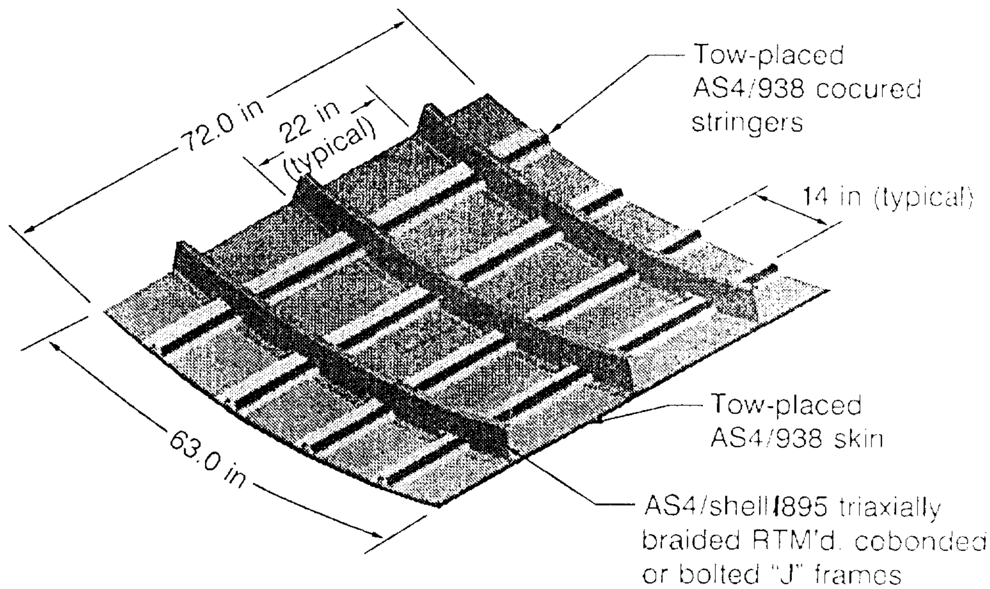

Figure 5. Fuselage crown test panel configuration.

PRESSURE BOX TEST FIXTURE CONFIGURATION

The pressure box was initially conceived as a fixture for the testing of stiffened panels of a fixed configuration. After discussions with NASA, and the creation of the Benchmark test program, the need for testing panels with differing configurations arose. Also the tension fracture work highlighted the need for some test data for curved panels, without frames and stringers, but with circumferential tear-straps representing frame flanges. The fixture configuration is not conducive to quick change over from one panel type to another. The requirement for axial loading necessitates attachment details and loading plates that have differing centers of gravity, or waterlines, so that for each panel axial loads are applied at the respective panel waterline. This must be achieved within reasonable limits so as to reduce any bending that may be applied to the test panel to a minimum. The use of pairs of load actuators at each end to apply axial load further complicates the set-up in that the actuators now must be on butt lines as well as waterlines due to the panel curvature. One modification, that may be made for the later Benchmark tests that NASA has scheduled for this fixture, is that of eliminating one axial load actuator at each end so that the differing panel butt line problem is removed.

Figure 6 shows an overall view of the pressure box fixture with the curved tear-strap panel installed. The dual axial load actuators can be seen at each end of the test panel, and the hoop load reactions on each side to react the internal pressure. Along each edge of the test panel are the individual double lap attachment fingers which apply axial load, or react the hoop loads from internal pressure. These attachments are individualized in order that they do not pick up transverse loads which would be diverted from the test panel. The test panels with frames and stringers, when installed, will also have individual hoop reaction members for each frame in addition to the skin reactions. These will be configured such that the frame stiffness is continuous for as far as practical through the hoop reaction systems. The weight of the attachment and loading plates is considerable, so a counterbalance system is used to ensure that this weight is not applied to the test panel. All of the test fixture components are fabricated from various steel and aluminum alloys, ranging from A-36 low strength steel for the pressure box and frame weldments, A-514 medium strength steel for the grip fingers and hoop attachment plates, 4340 high strength steel for the actuator clevis fittings and hoop turnbarrels, to 7075 aluminum for the tether straps and access doors in the pressure box itself.

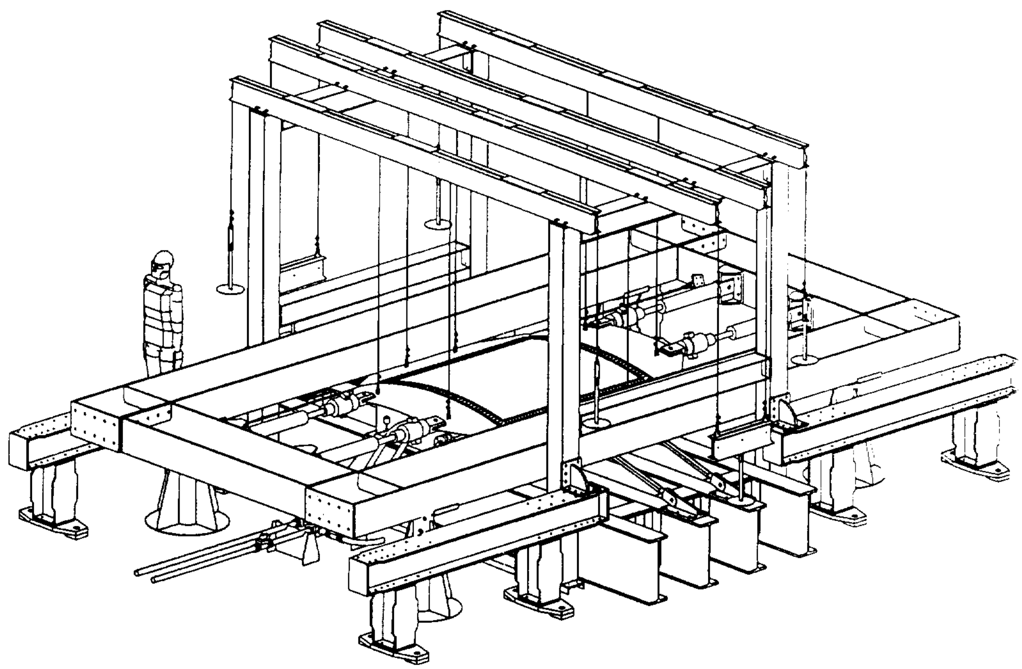

Figure 6. Pressure box test fixture.

Figure 7 presents a view of the fixture with the test panel removed so that the pressure seal and the interior of the pressure box may be seen. When the box is pressurized, the seal, being flexible, inflates slightly, thus causing the test panel to float, therefore the seal had to be designed such that no extraneous loads are applied to the test panel. The seal is fabricated from a fluoroelastomer/Kevlar laminate and is based on the advanced flexible tooling developed for the fabrication of the stiffened crown panels (Ref. 1). The seal is fabricated and autoclave cured in four sections, and bonded in each corner with a single lap shear splice. The single lap shear splice configuration was tested for strength prior to incorporation in the design. The seal is mechanically attached to both pressure box and test panel, and these attachment areas of the seal are reinforced with an aluminum strip to provide bearing strength. As the test panel floats when under internal pressure due to the seal inflating, a means of adjusting the alignment of the panel is necessary. This adjustment is provided by the turnbarrels in each of the hoop reaction systems. The overall size of the fixture is 25 feet in length, 20 feet in width, and 12 feet in height; this includes the overhead support beams for the counterweight system and shadow moire equipment.

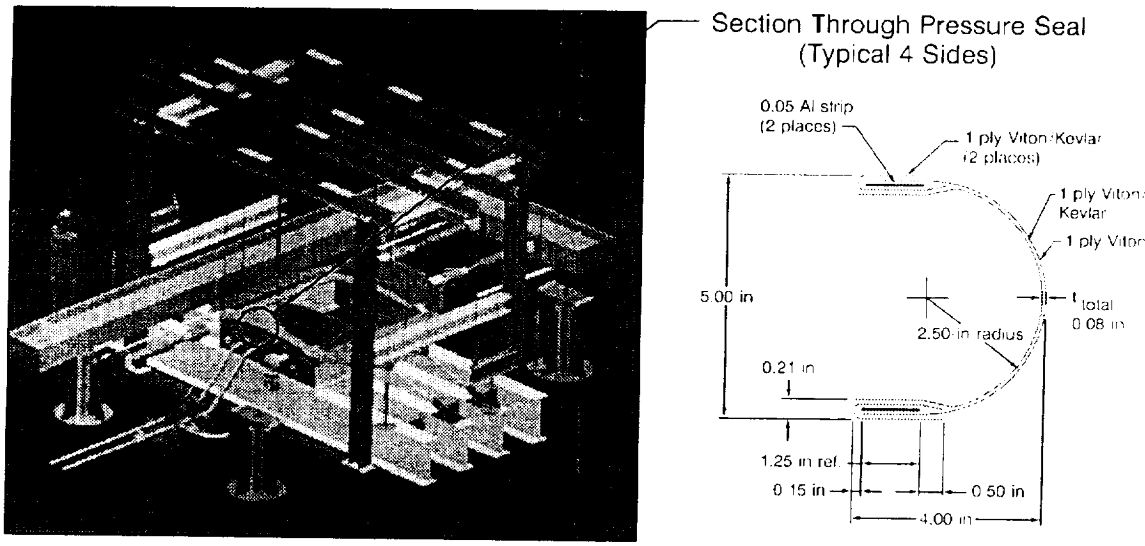

Figure 7. Pressure box seal detail.

PRESSURE BOX FIXTURE ANALYSIS AND DESIGN

A NASTRAN finite element model (FEM) was made of the pressure box fixture and test panel to help minimize the boundary effects of the fixture on the behavior of the test panel under load. As shown in Figure 8 only a quarter of the fixture and panel was included in the model due to symmetry. Another and important reason for the FEM analysis is to help understand the test results especially from panels with damage. From previous work (contract NAS1-17740, *Development of Composites Technology for Fuselage Structures in Large Transport Aircraft*) it was seen that the test panel dimensions needed to be very large if the effects of the fixture on the test panel are to be negligible, and this approach is impractical because of the increased costs of both test panels and the fixture.

A complete shell FEM was created at NASA Langley Research Center (LaRC) to support this work, and compared to a model of the test panel with symmetric boundary conditions, as well as the model of the complete test fixture and test panel. These FEM runs indicated that the fixture needed to be as stiff as possible in order to best approximate full fuselage shell boundaries for the test panel, so the attachment plates, individual grip fingers, hoop reaction members, and load plates were all changed to steel from aluminum. The FEM analyses also highlighted the need for hoop reaction nearer the fixture corners, so the reaction members were moved closer to the corners. A summary of the design revisions based on the FEM effort is shown in Figure 9.

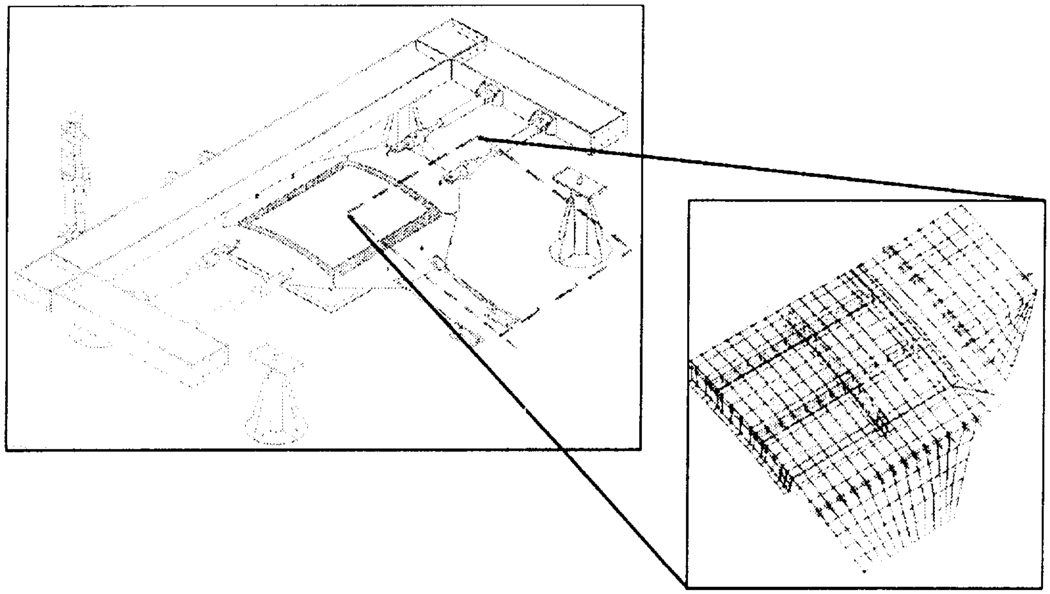

Figure 8. Finite element model of pressure box fixture and test panel.

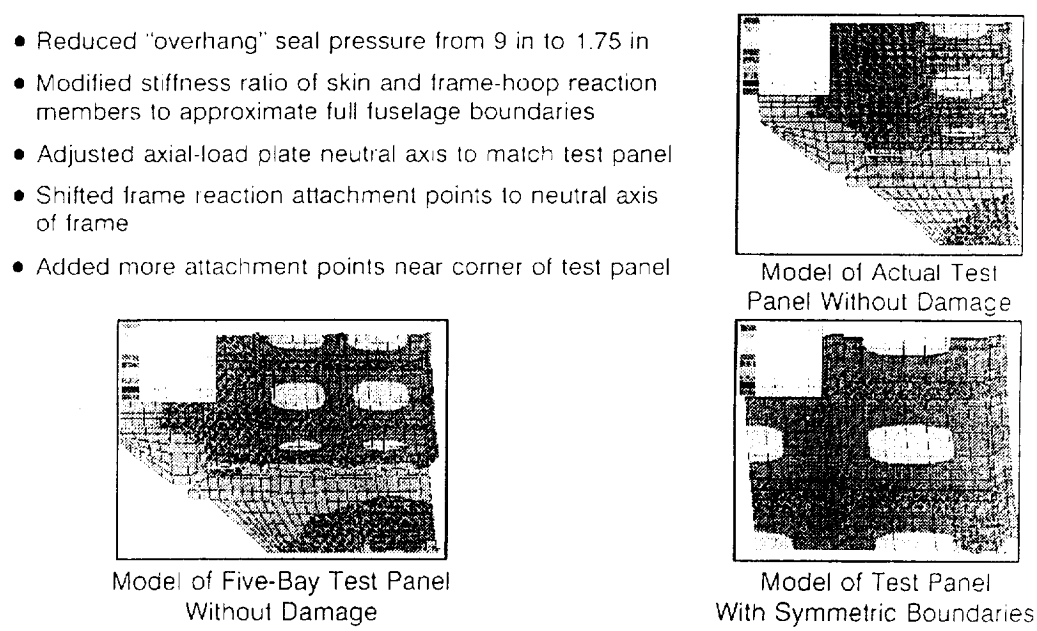

Figure 9. Design revisions to the pressure box fixture based on finite element modeling

The length of the stiffened test panel was a concern in that there are only three frame bays, and with the center frame and skin severed, just one intact frame bay on each side. The proximity of the fixture attachment plates could have a significant effect on the load redistribution around the ends of the damage in the center bay, thus making analysis of the test results difficult. Figure 10 presents the actual three bay test panel with a 22 inch skin notch and central frame severed under internal pressure loading of 10 psi. Figure 11 shows the test panel with an additional frame at each end with the same damage and loading as the three bay panel. There are differences in the stress field in the center bay, but these were not considered sufficient enough to warrant the additional length for the five bay panel.

Further evidence that analysis is absolutely necessary to be able to interpret the results of damage tolerance tests of configured panels in a pressure box fixture is presented in Figure 12. Figure 12 shows circumferential frame hoop loads plotted against distance from the shell crown centerline in degrees. These loads were extracted from full shell FEM analyses, performed by NASA LaRC personnel, and are those produced in the frame by a flight load case combined with 10.35 psi cabin pressure. The frame load plots are for a frame with and without severing damage. Also shown are the loads for the intact frames one and two bays away from the severed frame. It can be seen that when a frame is severed the axial loads in that frame in the vicinity of the damage are much disturbed, and do not become normal for a considerable distance, approximately 50 degrees from the damage. The shaded area of the figure indicates that portion of the shell that the pressure box test panels represent. Indeed it would require a pressure box test section measuring 9 feet arc length to allow the frame loads to be redistributed completely for a panel with a severed frame. The loads in the intact frame one bay forward are also much changed, and it can be seen that the pressure box would need to have a test panel that has five frames. This would indicate a total test panel length of almost 10 feet. The cost of such test panels would be on the order of three times as much as the current panels, with potentially a more costly fixture.

Not withstanding the limits of the test fixture, effective FEM analyses of the pressure box and test panels, and the full shell, together with sufficient testing, will enable engineers to gain an understanding of the damage tolerance of configured composite crown panels. This will aid in the design of future composite fuselage structure, such that testing of large fuselage sections may be conducted with confidence.

PRESSURE BOX TEST PROGRAM

The intended pressure box test program will be in three phases. The first phase, shown in Table 2, consists of tests conducted on test panels designed and fabricated by Boeing as part of the ATCAS contract. The second phase, shown in Table 3, will consist of tests conducted on test panels designed and fabricated by Boeing as part of a Task Assignment contract (NAS1-19349). The third phase, which will not be discussed here, will consist of panels designed and fabricated by Douglas as part of their ACT contract (NAS1-18862).

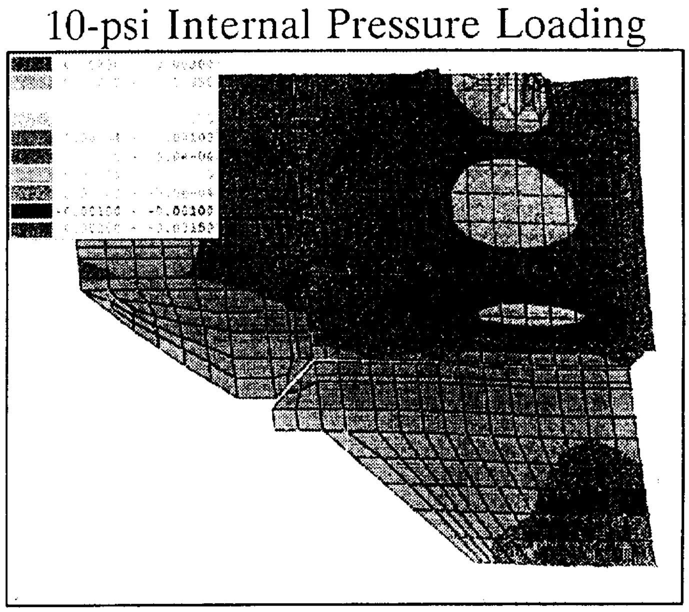

Figure 10. Finite element model of pressure box fixture and stiffened test panel with central damage.

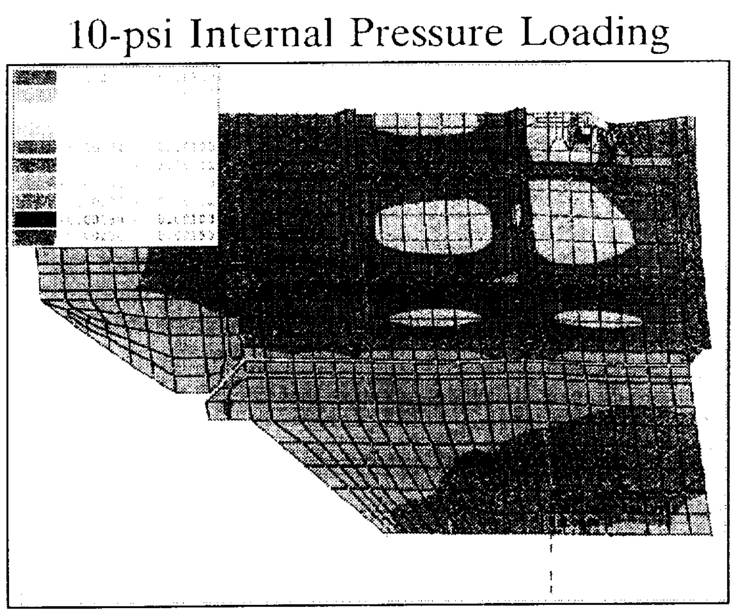

Figure 11. Finite element model of pressure box lengthened for a five-bay test panel with central damage.

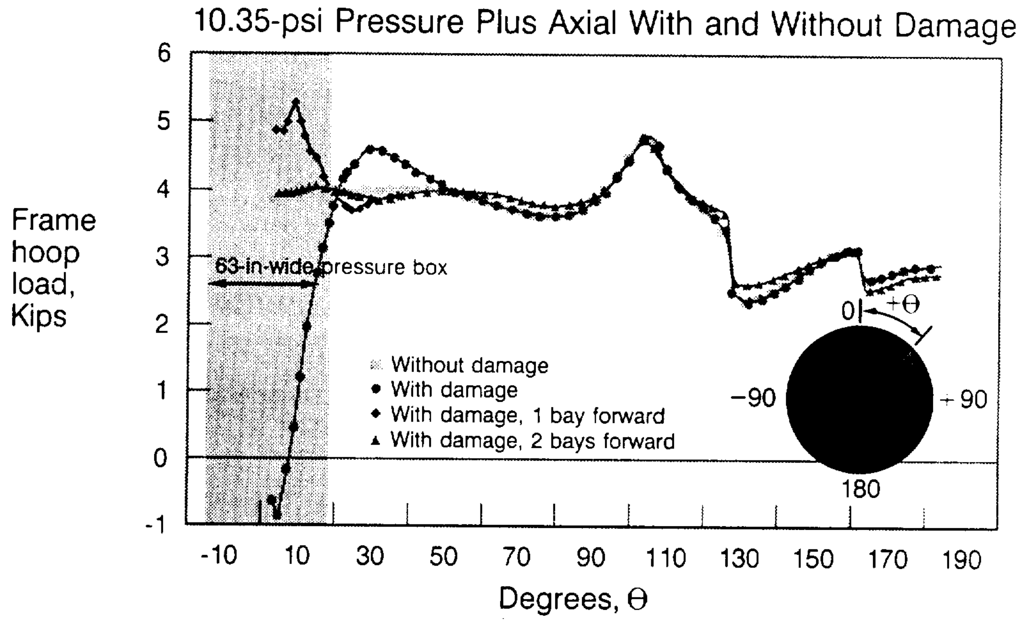

Figure 12. Circumferential frame hoop loads from full shell finite element analysis.

The first three panels in the phase 1 program will be tested at Boeing, the first being a tear-strap panel (reference Figure 4), the second and third being stiffened with frames and stringers (reference Figure 5). After the testing of the third panel is completed, the fixture will be disassembled, shipped to LaRC, and reassembled with Boeing coordination. The fourth panel, also stiffened with frames and stringers, will feature a repair of severing damage by American Airlines. The repair will be designed by Boeing based on the results of a current repair analysis and test program, with the actual repair performed by American Airlines personnel at their composite repair facility at Tulsa. This panel will be tested by NASA with Boeing coordination, after the pressure box fixture is moved from Boeing to LaRC.

Phase two of the test program will comprise the testing of five additional crown stiffened panels with varying details and damages. Table 3 lists all of these panels, their configurations, and testing scenarios. Both hoop tension critical damages and axial tension critical damages are included in the total program. Also hybrid skins will be featured on two of the phase two panels. The potential for enhanced tension fracture performance of hybrid panels has been demonstrated in ATCAS tension fracture work (Ref. 2). As was stated above, the design and fabrication of the panels for phase two are funded through a separate NASA contract. All of these panels will be tested at LaRC, with Boeing coordination.

Panel No.	Panel Description	Test Sequence	Panel test loads		Damage
			Internal Pressure	Axial Load [1]	
1	63" x 72" curved panel with 3 cobonded circumferential tear straps [2]	a) Strain survey with limit cabin pressure b) Strain surveys with increasing internal pressure	9.0 psi Up to 5.0 psi	380 lb/in —	a) None b) 22" longitudinal notch severing skin and central tear strap
2	63" x 72" curved panel with 4 cocured hat stringers and 3 bolted circumferential frames (min. gauge crown)	a) Strain surveys to limit cabin pressure b) Strain surveys with internal pressure only, then failure	9.0 psi 9.0+ psi	550 lb/in —	a) None b) 22" longitudinal notch, severing skin and central frame
3	63" x 72" curved panel with 4 cocured hat stringers and 3 cobonded circumferential frames (min. gauge crown)	a) Strain surveys to ultimate cabin pressure b) Strain survey to limit internal pressure only, then to failure	18.2 psi 9.0+ psi	1100 lb/in —	a) None b) 22" longitudinal notch, severing skin and central frame
4	63" x 72" curved panel with 4 cocured hat stringers and 3 cobonded circumferential frames (min. gauge crown)	a) Strain survey to ultimate cabin pressure b) Strain surveys to ultimate flight loads & cabin pressure	18.2 psi 13.5 psi	1100 lb/in 5000 lb/in	22" longitudinal notch, severing skin and central frame, repaired by American Airlines

[1] Includes representative longitudinal bulkhead pressure loading of PR/2.

[2] Predicted failure with damage of 5.0 psi internal pressure only

Table 2. Pressure box test program - Phase 1.

Panel No.	Panel Description	Test Sequence	Panel test loads		Damage
			Internal Pressure	Axial Load [1]	
5	63" x 72" curved panel with 3 cobonded circumferential frames and 4 cocured hat stringers	a) Strain survey up to limit cabin pressure and flight loads b) Load to failure with cabin pressure and flight loads	9.0 psi 9.0+ psi	550 lb/in 550+ lb/in	22" longitudinal notch, severing skin and central frame
6	63" x 72" curved panel with 3 cobonded circumferential frames and 3 cocured hat stringers (minimum gage crown area)	a) Strain survey to ultimate cabin pressure only b) Strain survey to limit cabin pressure and flight loads c) Strain survey to limit cabin pressure and flight loads, then failure	18.2 psi 9.0 psi 9.0+ psi	1100 lb/in 550 lb/in 3400+ lb/in	a) B.V.I.D.[2] on skin over central frame b) 2" central long. notch sev. skin at mousehole c) 14" circumferential notch, severing skin and central stringer
7	63" x 72" curved panel with 3 cobonded circumferential frames and 3 cocured hat stringers (Fwd. crown area)	Strain survey to limit cabin pressure and flight loads, then failure	9.0+ psi	5200+ lb/in	14" circumferential notch, severing skin and central stringer
8	63" x 72" curved panel with 3 bolted circum. frames and 4 cocured hat stringers (Hybrid skin - AS4/S2 fibers with 938 resin) fwd. crown	Strain survey to limit cabin pressure and flight loads, then failure	9.0+ psi	5200+ lb/in	30" circumferential notch, severing skin and two stringers
9	63" x 72" curved panel with 4 cocured hat stringers and 3 bolted circumferential frames (Hybrid skin - AS4/S2 fibers with 938 resin) min. gauge	a) Strain survey to limit cabin pressure b) Strain survey up to limit cabin pressure, then failure	18.2 psi 9.0+ psi	1100 lb/in —	a) None b) 22" longitudinal notch, severing skin and central frame

[1] Includes representative longitudinal bulkhead pressure loading of PR/2

[2] B.V.I.D. - Barely visible impact damage

Table 3. Pressure box test program - Phase 2.

TEST RESULTS

Phase 1 has commenced with testing of the first panel being completed. This panel, designated panel No. 1 in Table 2, was a curved tear-strap stiffened panel as presented in Figure 4. The test panel was instrumented with uniaxial and rosette strain gages, and deflection indicators. Shadow moire was used to provide a map of total panel deflections. The first test of the panel was a strain survey up to 9.0 psi internal pressure, with the panel undamaged. This first test run enabled the pressure box fixture and its systems to be checked out, and to understand how the panel was reacting to the internal pressure, comparing the strain and deflection to those from the FEM analysis. Axial load was applied to the panel, simultaneous with the internal pressure, to represent the bulkhead pressure present in transport fuselage cabins. The axial load applied was, in fact, lower than the load of 550 lb/in that correctly represents the bulkhead load associated with a limit cabin pressure of 9.0 psi. The test results indicated that the fixture was applying loads to the test panel correctly, and that the strains and deflections were similar to those predicted by the FEM analysis, except along the edges of the panel, especially in the corners. The FEM is not modeling these areas correctly, and more detail is needed in the model in order to match the fixture stiffness in the corners. The test panel was inspected for test induced damage. The inspection indicated that no damage had occurred.

The panel was then damaged with a 22 longitudinal inch central notch, severing the skin and the central tear-strap. The panel was again inspected with pulse echo in order to understand the complete damage to the panel prior to testing. Additional strain gages were cemented to both sides of the skin and tear-strap in the vicinity of both ends of the notch. A rubber seal was bonded to the inside of the skin, sealing the notch, and the panel was loaded with internal pressure up to 2.5 psi, combined with the corrected axial load of 150 lb/in. After this test run the panel was again inspected with pulse echo equipment to ascertain if damage growth had occurred. The inspection indicated that damage to the panel sustained by this loading sequence was minimal, and the panel was then loaded again. This time the loading was internal pressure alone. This represents a load condition consisting of cabin pressure combined with a flight maneuver that loads the fuselage crown with axial compression. This load combination is considered critical for minimum gage structure. An analysis of an unstiffened shell with the mechanical properties of the tear-strap skin and with a 22 inch longitudinal notch, indicated that catastrophic failure would occur at 5.0 psi internal pressure. Therefore, with the need to be able to inspect and dissect the panel after the test to compare with the results of various non-destructive evaluation (NDE) techniques, the panel was loaded to 4.5 psi and then unloaded. The pulse echo equipment indicated some damage growth at the ends of the notch, but more accuracy was needed. The flexural stiffness on the panel around the notch tips was measured using an advanced NDE technique that utilizes flexural wave dispersion. This technique is discused in Reference 3. Figure 13 presents some of the results of these flexural stiffness measurements. It can be seen that close to the notch tip, the panel flexural stiffness has been reduced significantly, with the stiffness increasing as the distance from the notch tip increases. This change in stiffness indicates, as did the pulse echo inspection, that some damage growth had occurred.

The panel will be removed from the fixture so that further inspection can take place. After these inspections from both sides of the panel, sections will be cut from the panel so flexural and axial stiffnesses can determined by mechanical test, and the results compared to those from the NDE equipment. In the mean time the fixture will be readied for testing of the stiffened panels.

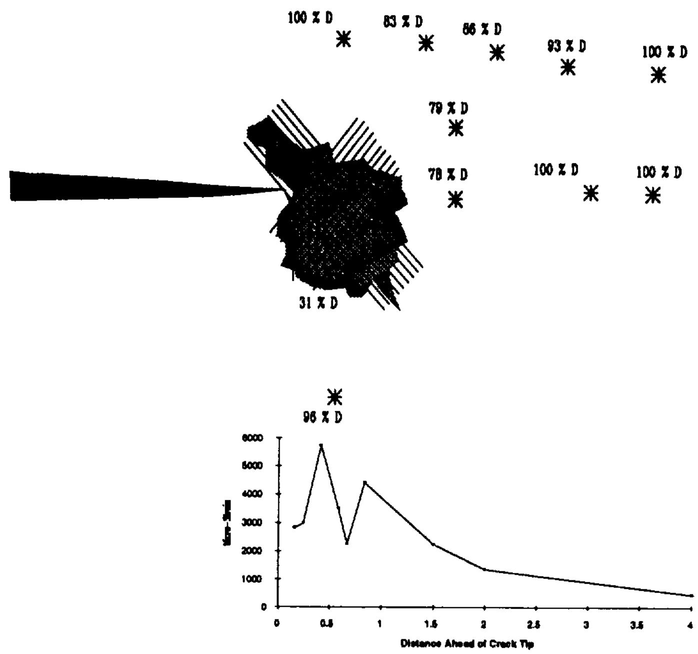

Figure 13. Effects of notch tip damage.

SUMMARY

In summary a pressure box test fixture has been designed and fabricated for the testing of curved fuselage panels. Analysis has aided considerably in the design process. The FEM analyses of both the fixture and test panels, together with the modeling work of full fuselage shells at LaRC, has resulted in a fixture that will be utilized by Boeing and NASA for tension damage tolerance testing of fuselage crown panels.

Testing has started with the first test completed. The pressure box fixture worked very well, in particular the flexible seal that was designed and fabricated in this work. As the testing continues with the stiffened panels, the analysis support at Boeing and LaRC will provide failure predictions and help understand the stress fields in the test panels. The tension damage tolerance data base that will result from both phases of the pressure box test program will help provide confidence for the effective application of advanced composites to commercial aircraft fuselage structures.

REFERENCES

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