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PERFORMANCE OF FUSELAGE PRESSURE STRUCTURE

James R. Maclin
Boeing Commercial Airplane Group

ABSTRACT

There are currently more than 1,000 Boeing airplanes around the world over 20 years old. That number is expected to double by the year 1995. With these statistics comes the reality that structural airworthiness will be in the forefront of aviation issues well into the next century.

This paper describes the results of previous and recent test programs Boeing has implemented to study the structural performance of older airplanes relative to pressurized fuselage sections. Included in testing were flat panels with multiple site damage (MSD), a full-scale 737 and 2 747s as well as panels representing a 737, a 777, and a generic aircraft in large pressure-test fixtures.

Because damage is a normal part of aging, this paper focuses on the degree to which structural integrity is maintained after failure or partial failure of any structural element, including multiple site damage (MSD), and multiple element damage (MED).

BACKGROUND

The 707, designed in the 1950s, was the first commercial jet airliner to be developed at Boeing with a pressurized fuselage. Experience at that time taught the aircraft industry that the ability to tolerate a substantial amount of damage was a requirement of modern airplanes. Therefore, the new pressurized fuselage had to be tested rigorously in order to prove airworthiness.

The test structure employed for design development and verification was made of large pressurized panels configured in the shape of a Quonset hut. For the tests, various combinations of skin gages and types of tear straps and shear ties were subjected to saw cuts and punctures by guillotine blades. Some early combinations proved inadequate as they resulted in dynamic destruction of the test panels. Eventually, several light-weight designs were verified to safely tolerate a high degree of damage (Figure 1).

At that time, MSD was not included in the Quonset testing, but was present in the full-scale fatigue tests of a production configuration.

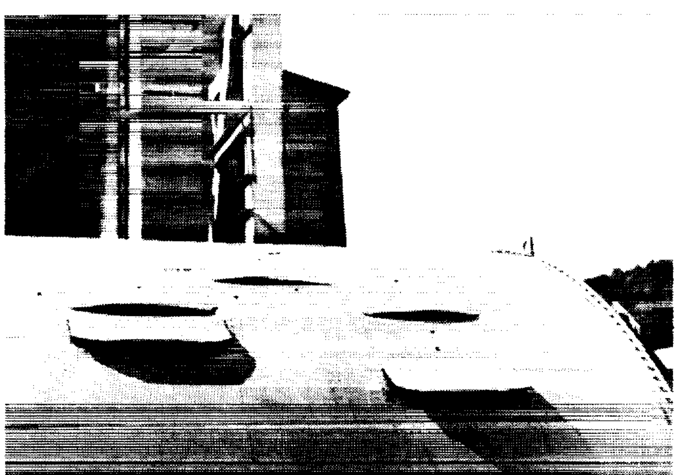

Figure 1. 707 Test Panel.

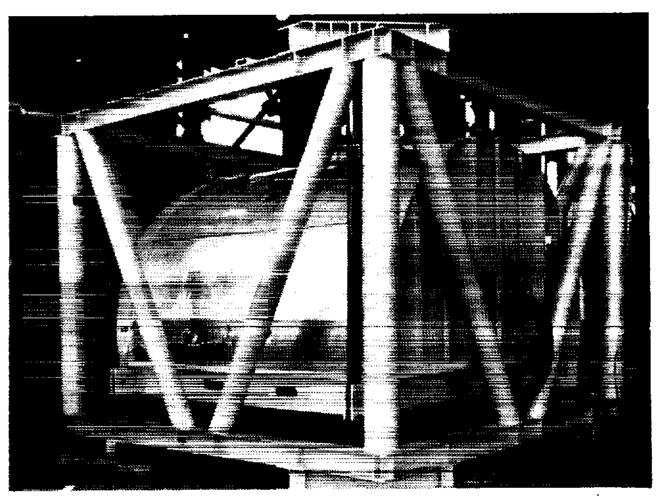

Figure 2. 737 Test Panel.

During full-scale fatigue tests, cracks developed in lap splices with many cracks at fasteners in adjacent bays. These were natural fatigue cracks (MSD) that grew until they eventually linked up in one bay, then formed flaps that allowed safe decompression.

Structural designs that incorporate tear straps and/or shear ties under the fuselage frames at 20-inch spacing were verified by guillotine and fatigue tests and subsequently also verified in service by the 707, 727, 747, 757, and 767.

Special tests were devised for the 737, which was a lighter weight. Because of lower operating cabin pressure and lower body bending loads due to its shorter fuselage, the 737 was designed with a thinner 0.036-gage skin.

Again, a Quonset hut test panel was used for development, except the test fixture for the 737 was loaded with skin shear in addition to internal pressure (Figure 2).

The final design for the 737 incorporated a waffle doubler bonded to the 0.036-gage skin with tear straps at 10-inch intervals.

Dual 12-inch-wide guillotine blades straddling a frame were used to test its structural damage tolerance (Figures 3 and 4). The results of these tests are shown in Figure 5.

Figure 3. Double Guillotine Blade Test.

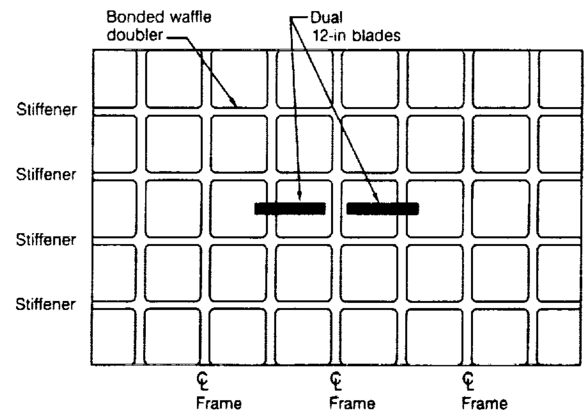

Figure 4. 737 Developmental Guillotine Test Panels.

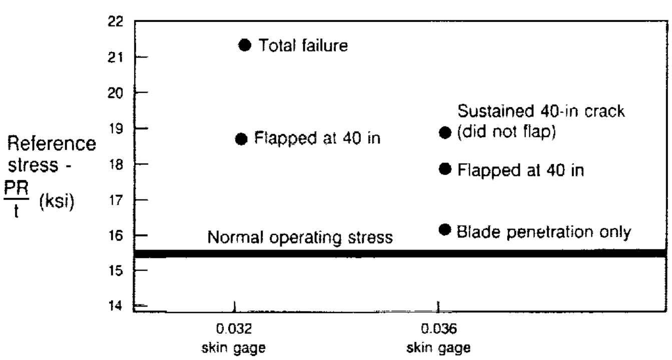

Figure 5. 737 Developmental Guillotine Test Results.

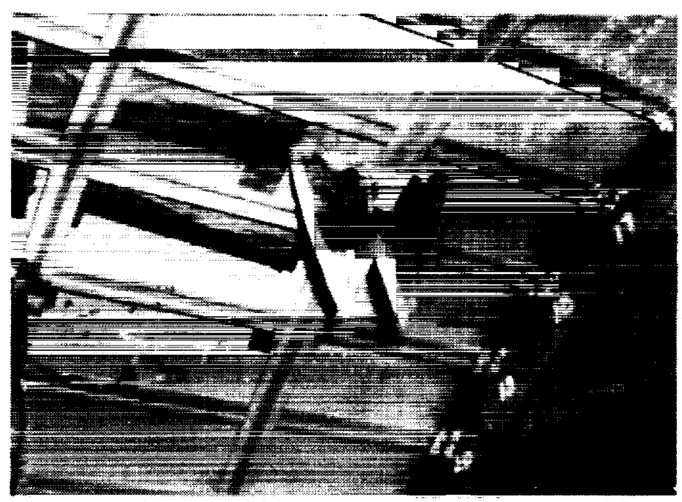

Figure 6. 737 Guillotine Test.

The tests demonstrated the capability of the 737 pressure structure to safely tolerate a substantial amount of damage, frequently cracks up to 40 inches long (Figures 6 and 7).

Saw cut and guillotine tests were also conducted on the 747 design. The use of tear straps and shear-tie designs created a fail-safe capability for cracks of at least one bay, or 20 inches long.

The primary difference noted between these tests and previous 707 and 737 tests was the tendency of the heavier 0.063-gage skin not to form flaps.

The designs of the 757 and 767 fuselage sections were essentially an improved version of the 707. During verification on a full-scale fuselage and after fatigue testing to twice the design objective, the designs demonstrated the ability to sustain two-bay cracks with a severed middle frame. Lighter skin gages, i.e., 0.040, on the 757 allowed safe decompression by flapping.

The design of the 727 fuselage section was so similar to the 707 design that test data were applicable to both and no additional testing was necessary.

MULTIPLE SITE DAMAGE CONSIDERATIONS

In the 1980s, with many airplanes in the world fleet approaching or exceeding their original design service objectives, the focus of the industry turned to damage scenarios associated with fatigue cracking.

Tests using saw cuts and guillotine blade punctures demonstrated a high level of damage containment whether the source of the damage was accidental or due to local fatigue. However, widespread MSD fatigue cracking at many locations raised new questions.

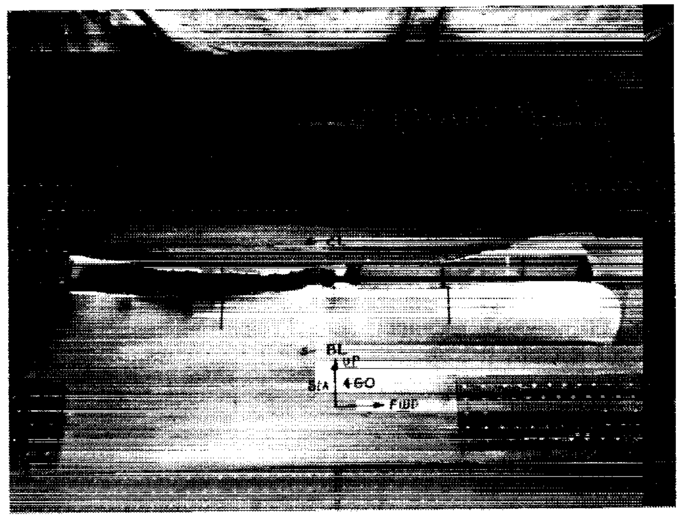

Figure 7. Example of Safe Two-Bay Flapping.

Boeing had relied on data produced by 707 fatigue tests that demonstrated safe decompression in lap splices when natural cracking occurred in several adjacent bays, but they were uncertain what would happen if the natural MSD was more severe than had occurred due to fatigue scatter. Also, more information was needed for heavier skin gages that tended not to form flaps.

To address these concerns, Boeing initiated several test programs:

- Flat panel tests using flat panels, rather than the more elaborate Quonset hut, to obtain an early indication of the probable effects on residual strength of a common scenario—small fatigue cracks remaining in a row of fastener holes after some cracks have linked up to become a large crack.
- Fatigue tests on a 737 aft fuselage section from an aircraft previously in service.
- Fatigue tests on a 747-100SR fuselage section from an aircraft previously in service.
- Fatigue tests on a new 747-400 forward fuselage.
- Construction of two large fuselage pressure-test fixtures and testing of a variety of 737, 777, and generic test panels.

New data obtained from these test programs were examined to determine the effects of MSD on the damage tolerance performance of fuselage pressure structure.

FLAT PANEL TESTING

Natural fatigue cracking with MSD frequently involves small cracks at fastener holes in a row preceding a larger crack in the same row. Flat panels were used in this test program as a quick method for testing residual strength, while avoiding the time of loading curved, more realistic fuselage panels. For this reason, data obtained by this method were regarded as approximations only. The following three test locations were selected for flat panel testing (Figure 8):

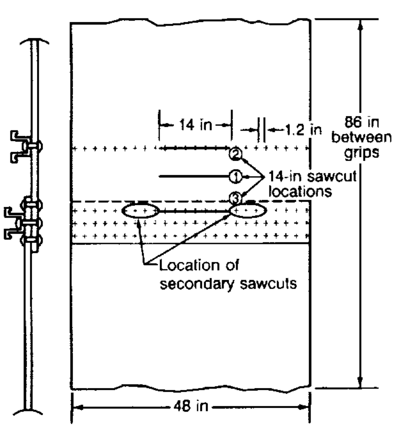

Figure 8. Flat Panel Test Specimen.

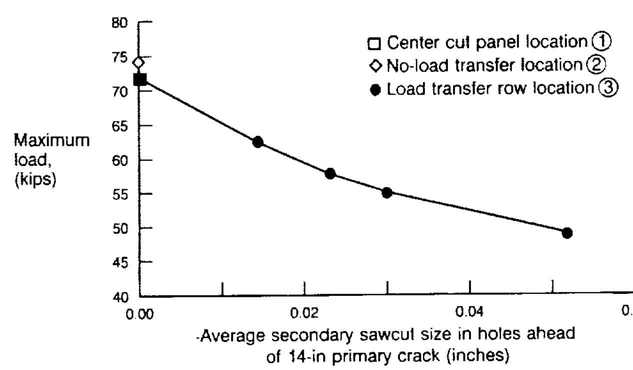

Figure 9. Flat Panel Test Results.

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Location 1: A baseline for testing material properties involving a saw cut, but without a row of holes preceding the cut.

Location 2: A row of holes in an area where no load is transferred to test for the effect of holes without MSD.

Location 3: The top row of a lap splice to include the effects of both fastener holes and load transfer; also used to determine the effect of small fatigue cracks, simulated by saw cuts approximately 0.02 inches wide, on the residual strength of the panel containing a large crack.

Test results are shown in Figure 9. The graph indicates that there was no significant difference in residual strength between locations 1, 2, and 3 until the small MSD saw cuts were added to location 3. At that point, the amount of load tolerated rapidly declined as the size of the small saw cuts increased. Because of this significant drop, the phenomena will be investigated further under more realistic conditions using a pressure-loaded test fixture.

737 AFT FUSELAGE FATIGUE TEST

Another step in evaluating the effect of MSD on damage tolerance was the fatigue testing of a retired 737 aft fuselage (Figure 10). After 59,000 service flights, the fuselage test section (Figure 11) was cycled until normal fatigue cracking had begun and grown to its natural conclusion of a two-bay crack with safe decompression by flapping (Figure 12).

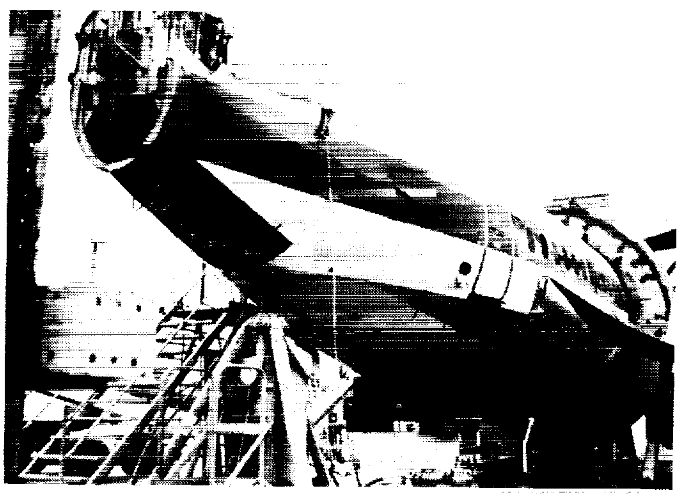

Figure 10. Retired 737 Aft Fuselage Fatigue Test Article.

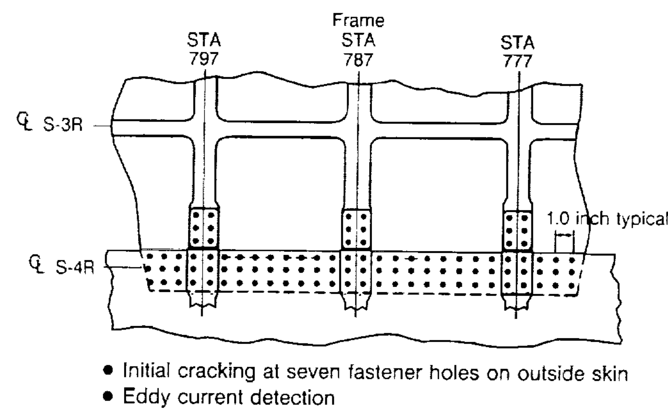

Figure 11. Initial 737 Lap Splice Cracking.

In this test, MSD was present in adjacent holes and in adjacent frame bays with a nonuniform distribution of crack sizes typical of fatigue scatter. Although the test may not represent the worst-case of fatigue scatter and MSD, it is reasonable to assume that the test results do represent typical performance.

Figure 12. Controlled Decompression of 737 Fuselage Test Article.

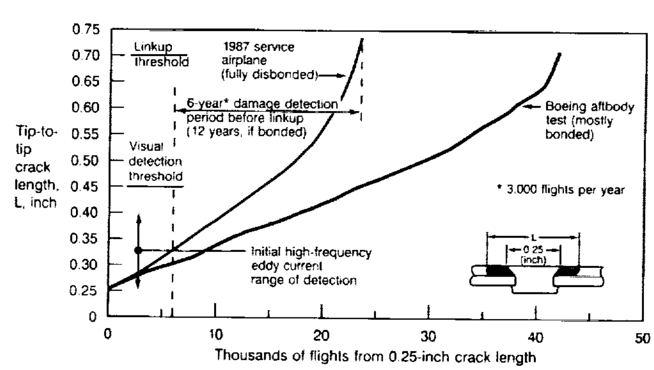

Figure 13. 737 Lap Splice Crack Growth Data.

Decompression by flapping is not relied on as a safety factor in the case of cracks in lap splices. Rather, inspection programs are in effect that ensure crack detections before linkup or very soon thereafter. In the test described above, a 12-year damage detection period between initial detection and linkup was indicated assuming 3,000 flights per year (Figure 13). According to further experience on a fully disbonded inservice airplane, that number would be reduced to 6 years—still ample time for detection.

747-100SR FATIGUE TEST

For this test, a 747-100SR with an equivalent of 20,000 full-pressure cycles (flights) was obtained from service and monitored as it was extended to 40,000 cycles (Figure 14). The minimum gage lap splices are shown in Figure 15.

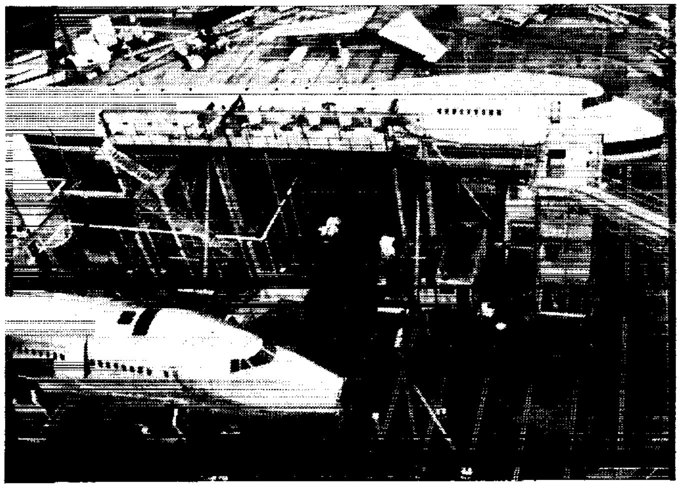

Figure 14. Retired and New Model 747 Fuselage Test Articles.

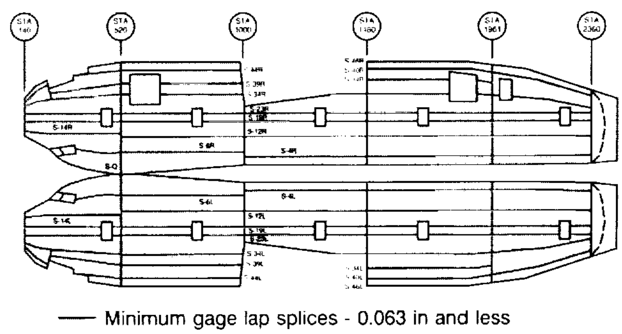

Figure 15. 747-100SR Fuselage Fatigue Tests.

An initial crack was detected in the lap splice in S-14R at 21,500 cycles. This crack eventually linked up with other small cracks and grew to approximately 6 inches by the time the test was concluded at 40,000 cycles (Figure 16).

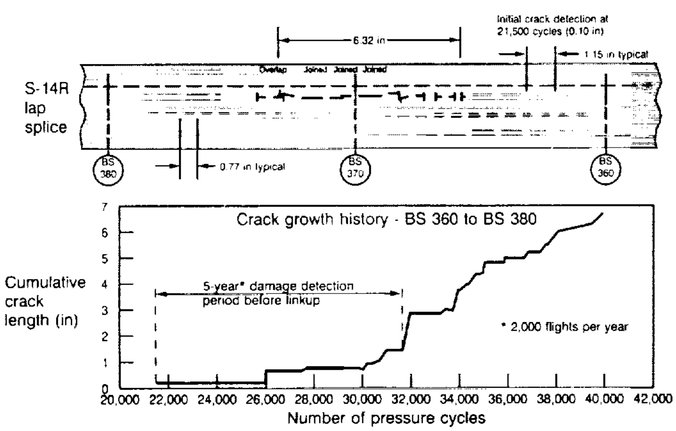

Figure 16. 747-100SR Fuselage Fatigue Test Results.

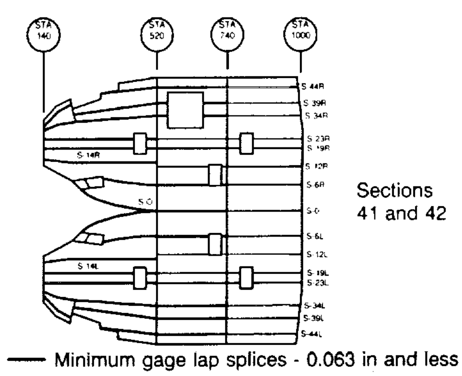

Figure 17. 747-400 Fuselage Fatigue Tests.

Assuming 2,000 flights per year of normal operations, the crack growth data from this test indicate a 5-year damage detection period before linkup. After linkup, tests indicate there is a significant, additional safe damage detection period.

747-400 FATIGUE TEST

Sections 41 and 42 from the 747-400 production line were pressure cycled to determine some of the fatigue and damage tolerance characteristics of the latest production configurations (Figure 14). The minimum gage lap splices are shown in Figure 17.

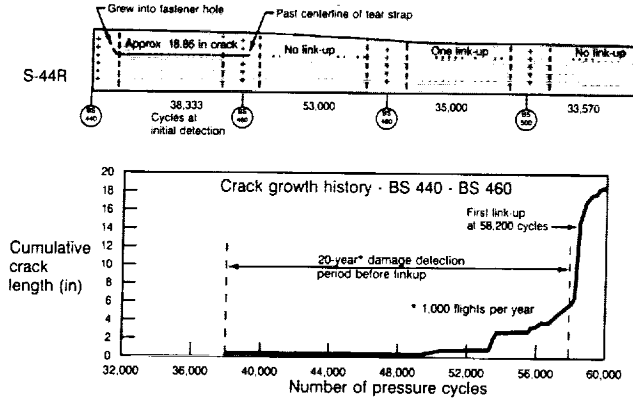

Figure 18. 747-400 Fuselage Fatigue Test Results—S44R

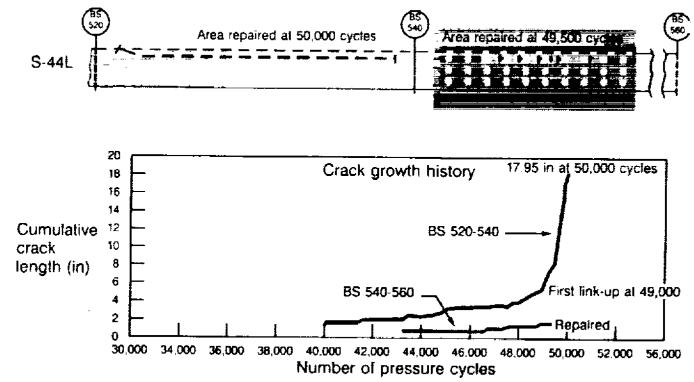

Figure 19. 747-400 Fuselage Fatigue Test Results—S44L.

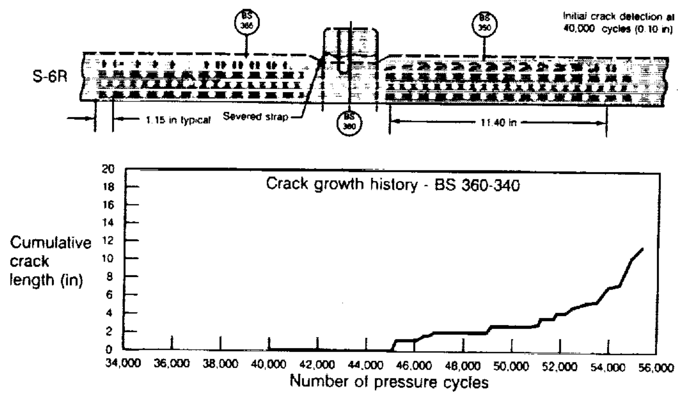

Figure 20. 747-400 Fuselage Fatigue Test Results—S6R.

Cracks eventually started at several locations, but appeared later and grew slower than similar cracks on the 747-100SR (Figures 18, 19, 20). These crack growth data indicate a long damage detection period between the time of detection and linkup, possibly as long as 20 years for an aircraft making 1,000 normal operation flights per year.

The data also show that the 747-400 fuselage section is capable of supporting a one-bay crack, providing an additional safe crack detection period.

PANEL TESTING IN LARGE PRESSURE-TEST FIXTURES

In 1989 and 1990, Boeing built two large pressure-test fixtures, one with a 74-inch radius, representing a narrowbody, and one with a 127-inch radius, representing a widebody (Figure 21). These fixtures were designed to accommodate testing for fatigue, crack growth, and residual strength of large pressure panels with a variety of structural designs and details.

A typical test panel configuration is shown in Figure 22; however, up to three lap splices can be accommodated and test locations can vary. Test panel frame spacing, stringer spacing, and panel radius are set by the fixture.

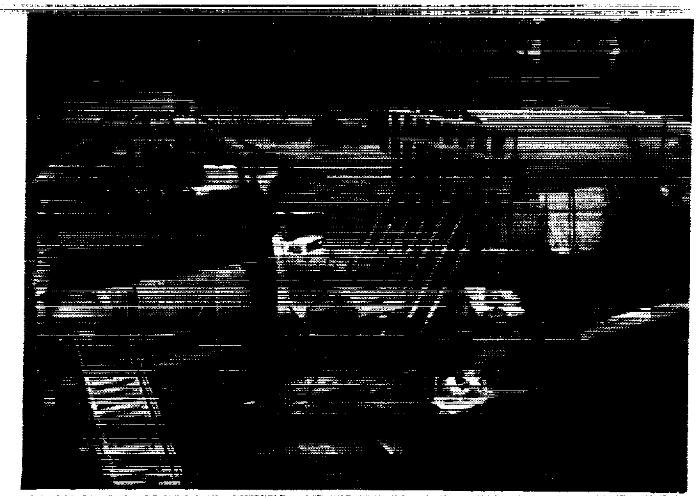

Figure 21. Boeing Pressure-Test Fixtures.

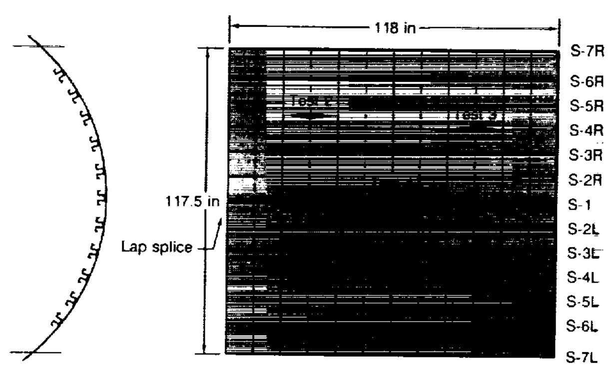

Figure 22. Typical Pressure-Test Panel.

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Although attention had been given to a structural fuse system, damage was sustained during the first test when the test panel reached dynamic failure. Since then, the fixture has been repaired and the fuse arrangement modified.

To date, the new system has protected the fixtures from further damage during six dynamic panel failures, three on each test fixture. Damage tolerance testing has been done on 737 panels, 777 developmental panels, and on generic panels (Figures 23, 24, 25 respectively).

The 737 tests on the large pressure-test fixtures represented panel production configurations and field repair of a disbonded waffle doubler. Four of the tests were in lap joints. From these data, three major observations were derived:

Panel configuration - narrow body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Dynamic at 17% overpressure
• Riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Riveted repair of tear straps Centered on frame	1 bay (~20 in)	Flapped
• Bonded tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Bonded tear straps Centered on frame	1 1/2 bays (~30 in)	Flapped
• Blind riveted repair of tear straps Centered mid bay	1 bay (~20 in)	Flapped
• Blind riveted repair of tear straps Centered on frame	1 1/2 bays (~30 in)	Flapped
• Lap joint Riveted repair of tear straps	2 bays (~40 in)	Flapped
• Lap joint with simulated MSD Riveted repair of tear straps	1 bay (~20 in)	Dynamic
• Lap joint with broken central frame Riveted repair of tear straps	1 bay (~20 in)	Dynamic
• Lap joint with simulated MSD Bonded tear straps	1 bay (~20 in)	Dynamic

Figure 23. Damage Tolerance Testing—737.

- All panels were able to sustain at least a one-bay crack at operating pressure, including those with small saw cuts in the lap joints, simulating conservative MSD.
- Damage at the lap joint location, which produced safe decompression by flapping without MSD, caused dynamic failure after a one-bay crack when a generous degree of MSD was added.
- All tests conducted outside the lap splice provided safe decompression by flapping, except one panel that was overpressurized by 17%. That panel failed dynamically.

The 777 test panel configurations were experimental and may not represent the design actually selected for production. All panel configurations tested were fully shear-tied. The primary differences between panels were in the materials used; small changes in thickness of skin, shear ties, and frames; and a few other details.

The data showed that all panels were able to sustain operating pressure with two-bay cracks or greater, with the central frame severed (Figure 24). The latest panel tested as of this writing held a 20% overpressure without failure. One panel was tested with a saw cut in the lap joint without MSD and performed in approximately the same manner as the non-lap splice locations.

Panel configuration - wide-body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Broken central frame	2 bays (~40 in)	Repaired
• Broken central frame	Over 2 bays (~58 in)	Dynamic
• Lap joint - broken central frame	Over 2 bays (~48 in)	Repaired
• Broken central frame	2 bays (~40 in)	Repaired
• Broken central frame	Over 2 bays (~55 in)	Dynamic
• Broken central frame	2 bays (~40 in)	Held 20% over pressure

Figure 24. Damage Tolerance Testing—777 Development.

Panel configuration - wide-body pressure test fixture	Maximum crack length sustained at operating pressure	Residual strength results
• Bonded tear straps - 0.071 in skin Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Crack between stringers	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps - 0.063-in skin Lap splice with simulated MSD and broken frame	2 bays (~40 in)	Held 9.5 psi prior to cutting frame Dynamic at 9.0 psi with cut frame
• Shear ties and tear straps Crack near stringer	1 bay (~20 in)	Repaired
• Bonded tear straps Crack between stringers	1 bay (~20 in)	Repaired
• Bonded tear straps Crack near stringer	1 bay (~20 in)	Repaired
• Lap splice with simulated MSD and broken frame Shear ties and tear straps	2 bays (~40 in)	Dynamic at 9.4 psi

Figure 25. Damage Tolerance Testing—Generic.

The two generic pressure-test panels were each tested at four locations (Figure 25). The first three locations were tested to collect load redistribution data from strain gages and panel failure was not expected. The focus of tests on both of these panels was the residual strength of the lap splice with MSD.

Small saw cuts, with a distribution of depths up to 0.020 inch, were made on the center four frame bays prior to panel assembly. Each panel sustained at least a two-bay crack in the lap splice with the central frame severed at normal operating pressure. The panels were then overpressurized until failure.

CONCLUSIONS

The following conclusions were made:

- Guillotine tests, started in the 1950s, have shown that pressure structure reinforced with tear straps and/or shear ties can safely sustain a large amount of damage at any location.
- At locations where fatigue cracks are likely to develop in a row of fastener holes, small cracks under 0.03 inch at all holes can significantly reduce the residual strength of a large crack in the same row of holes.
- All configurations tested, including those with MSD in adjacent holes, show the structure can safely contain at least a one-bay crack of approximately 20 inches. Configurations with at least 19% stiffening tear straps and/or full shear ties can contain a two-bay skin crack with a severed central frame. However, widespread fatigue damage in adjacent bays could reduce these critical size cracks.
- Although all testing has shown that typical fuselage pressure structure can sustain at least a 20-inch crack, the best opportunity for safely detecting a crack in a longitudinal skin splice is before the crack reaches 1 to 2 inches. Detecting cracks in their early stages takes best advantage of the long time period (5 to 20 years) between detection and linkup. Also, as the crack becomes longer, the likelihood of widespread fatigue cracking in adjacent structure increases, resulting in reduced residual strength.
- Cracks in skin gages of 0.040 or less, reinforced with tear straps and/or shear ties, show a strong tendency to form flaps and provide safe decompression, except when the cracks appear in a row of fasteners containing a large (and perhaps unrealistic) amount of MSD. Cracks in skin gages 0.63 inch and greater have not demonstrated this tendency to form flaps.

SUMMARY

Thirty-five years of testing for performance of fuselage pressure structure have now been reviewed. Saw cut and guillotine tests have shown a substantial and adequate amount of damage can be contained when the skin of an aircraft is reinforced with tear straps and shear ties, as is standard at Boeing and typical of the industry today.

These data show further that current standards are satisfactory for areas of the fuselage pressure structure and are not susceptible to widespread MSD. However, the fuselage longitudinal splices are susceptible to widespread MSD and need special consideration.

The 737 panel testing has demonstrated that conservative amounts of MSD may prevent the formation of flaps and, therefore, prevent safe decompression. In addition, flat panel tests indicate that MSD may reduce the residual strength for heavier skin gages that do not tend to form flaps.

The 707 full-scale fatigue test and the 737 aft fuselage fatigue test have shown that safe decompression occurs by flapping and that naturally-generated MSD did not prevent flapping. The concern is that a worst-case MSD due to normal fatigue scatter could prevent flapping and could also reduce the residual strength for heavier skin gages that do not tend to flap. These concerns dictate that an inspection program is required to detect cracks before they start to interact with cracks in adjacent frame bays.

Examination of the crack growth data from the 737 aft body test and service, the 747-100SR test, and the 747-400 test indicates a damage detection period of 5 to 20 years between detection (using current practical technology) and linkup or near linkup. Further, an additional safe damage detection period exists after linkup. The overall time period available to safely detect cracks in longitudinal joints is adequate provided the necessary inspections are identified and performed.

RECOMMENDATIONS

In light of the above tests, the following recommendations are made:

- Even in cases where an adequate damage detection period is available, structural modification should be considered as an alternative to an intensified inspection program after the structure has reached a service threshold for possible MSD initiation. This modification has the advantages of eliminating the risk of the inspections not being adequately performed and providing timely scheduling of a "fix," which is likely to become necessary.
- To address worst-case scenarios, service thresholds for possible MSD initiation and the subsequent damage detection period should be based on the disbonded condition of longitudinal fuselage splice areas. Disbonds in lap splices have occurred in service resulting in significant loss of fatigue performance and faster crack growth rates.