

EVALUATION OF THE FUSELAGE LAP JOINT FATIGUE
AND TERMINATING ACTION REPAIR

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SUMMARY

Terminating action is a remedial repair which entails the replacement of shear head countersunk rivets with universal head rivets which have a larger shank diameter. The procedure was developed to eliminate the risk of widespread fatigue damage (WFD) in the upper rivet row of a fuselage lap joint.

A test and evaluation program has been conducted by Foster-Miller, Inc. (FMI) to evaluate the terminating action repair of the upper rivet row of a commercial aircraft fuselage lap splice. Two full scale fatigue tests were conducted on fuselage panels using the aircraft panel test facility at FMI. The first was conducted to characterize initiation and growth of fatigue cracks in the lap joint. The second test was performed to evaluate the effectiveness of the terminating action repair. In both tests, cyclic pressurization loading was applied to the panels while crack propagation was recorded at all rivet locations at regular intervals to generate detailed data on conditions of fatigue crack initiation, ligament link-up, and fuselage fracture.

This program demonstrated that the terminating action repair substantially increases the fatigue life of a fuselage panel structure and effectively eliminates the occurrence of cracking in the upper rivet row of the lap joint. While high cycle crack growth was recorded in the *middle* rivet row during the second test, failure was not imminent when the test was terminated after cycling to well beyond the service life. The program also demonstrated that the initiation, propagation, and linkup of WFD in full-scale fuselage structures can be simulated and quantitatively studied in the laboratory.

This paper presents an overview of the testing program and provides a detailed discussion of the data analysis and results. Crack distribution and propagation rates and direction as well as frequency of cracking are presented for both tests. The progression of damage to linkup of adjacent cracks and, to eventually, overall panel failure is discussed. In addition, an assessment of the effectiveness of the terminating action repair and the occurrence of cracking in the middle rivet row is provided, and conclusions of practical interest are drawn.

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BACKGROUND

The problems associated with fatigue were brought to the forefront of research by the explosive decompression and structural failure of the Aloha Airlines Flight 243 in 1988. The structural failure of this airplane has been attributed to debonding and multiple cracking along the longitudinal lap slice in the fuselage [1].

The Federal Aviation Administration Technical Center (FAATC) has initiated several research programs to investigate the structural integrity of aging airplanes. Particular attention has been given to understanding the phenomenon of multiple cracking. The term "Widespread Fatigue Damage" (WFD) is commonly used to refer to a type of multiple cracking that degrades the damage tolerance capability of an aircraft structure. One of the issues regarding WFD is to determine its onset, or the point in time when the presence of multiple cracks are of sufficient size and density whereby the structure will no longer meet its damage tolerance requirement. Foster-Miller, Inc., under contract with the John A. Volpe National Transportation Systems Center, has conducted several test programs to support these investigations [2,3,4].

This paper summarizes the results from fatigue tests performed on two full-scale fuselage panels. A unique test facility, specially designed and built by Foster-Miller, Inc., was used to conduct these tests. A description of this test facility can be found in References [2] and [3]. The objectives of these fatigue tests are:

- (1) to characterize lap joint fatigue of an aircraft fuselage.
- (2) to study the initiation, growth, and linkup of multiple cracks.
- (3) to evaluate the effectiveness of the terminating action repair.

The first fatigue test addresses the first two objectives, and is regarded as a baseline test. In this paper, it is also referred to as the fatigue characterization test. A second test on a second full-scale panel was performed to address the last objective. Specific details of these full-scale fatigue tests can be found in Reference [4].

PANEL DESIGN

The basic panel configuration was developed under previous test programs conducted by Foster-Miller [2,3]. The panel was designed to be representative of older commercial aircraft with skin lap construction. Figure 1 shows a photograph of the underside of the test panel.



Figure 1. Photograph of test panel underside.

Specific key features of this panel include the following:

- (1) The panel material is 2024-T3 alclad aluminum alloy. The thickness of each skin is 0.036 inch.
- (2) Stringer ties were employed to join the stringers to the frames.

Additional panel features and dimensions are listed in Table 1.

Due to manufacturing and cost limitations, the test panel deviates in some design features from an actual commercial aircraft. While the fabrication process is simplified, these modifications do not impact structural performance of the fuselage panels. The most significant modifications include the following:

- (1) A tear strap and filler strip arrangement was used in place of the waffle doubler design found in actual aircraft. Typical aircraft construction consists of bonding two pieces of skin together and then chemically removing all the material which is not a tear strap or does not lie over a stringer. Thus, a continuous waffle pattern doubler is produced beneath the skin. In the test panels, continuous tear straps were bonded to the skin and longitudinal filler strips were used over the stringers.

- (2) The countersunk rivets were replaced with larger universal head rivets to prevent any unrepresentative cracking at the non-lap joint locations of stringers. In actual aircraft, the waffle pattern doublers over the stringers are hot bonded, eliminating the knife edge crack initiation site.

Table 1. Fatigue Panel Characteristics

| | |
|-------------------------------|-------------------------------|
| Panel length (inches) | 120 |
| Panel width (inches) | 68 |
| Panel radius (inches) | 75 |
| Number of frames | 6 |
| Number of tear straps | 11 |
| Number of stringers | 6 |
| Frame spacing (inches) | 20 |
| Tear strap spacing (inches) | 10 |
| Stringer spacing (inches) | 9.6 |
| Skin thickness (inch) | 0.036 |
| Tear strap thickness (inches) | 0.036 |
| Skin and tear strap material | 2024-T3 Aluminum alloy (clad) |

FATIGUE CHARACTERIZATION TEST

The pressure range for both test panels was between 1.0 and 9.5 psi. The pressure differential of 8.5 psi was used to account for the effects of aerodynamic suction and transverse shear due to body bending in addition to nominal cabin pressure. The rate of loading was 0.2 Hertz or 720 cycles per hour.

Cracking in the upper row of rivets was first observed after 37,000 cycles. Figure 2 shows the distribution of cracks along the upper rivet row at different points in time as the panel was cycled. The abscissa represents the rivet location where rivets have been numbered sequentially from the left to right on the test panel.

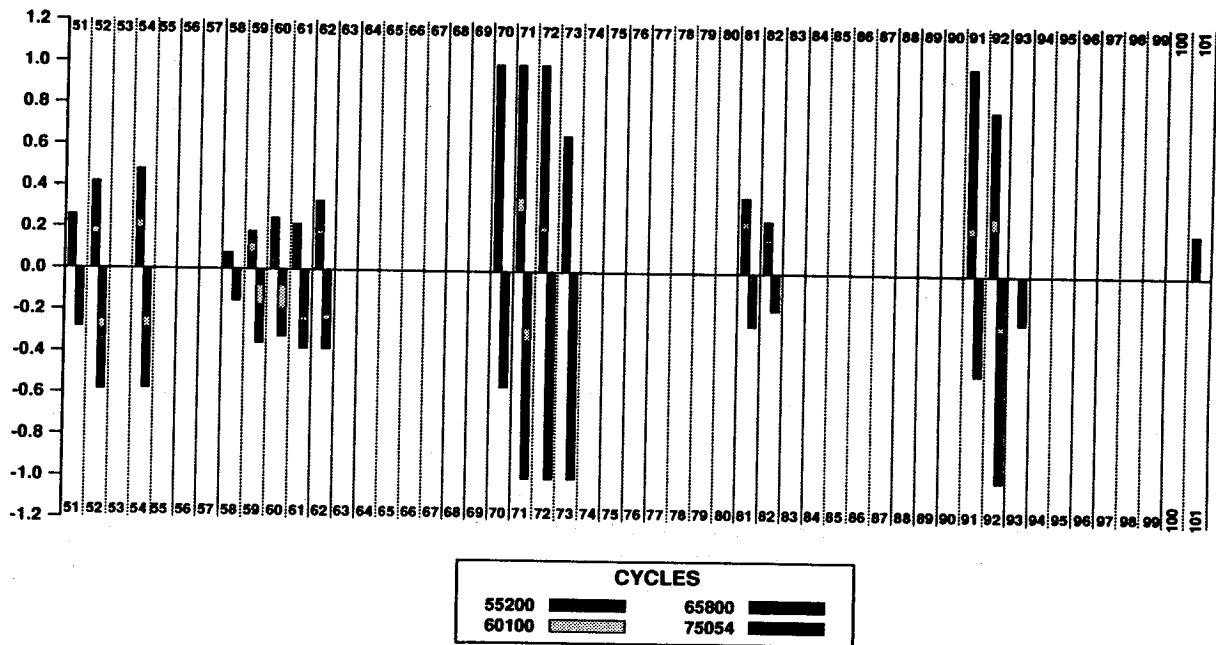


Figure 2. Distribution of cracks in the upper rivet row.

Figure 3 shows the sequence of multiple crack linkup events which led to failure of the panel. The first linkup of multiple cracks was observed after 69,400 cycles in the ligament between rivets 71-72. Similar linkups occurred in ligaments 91-92 after 73,325 cycles later, and in ligaments 46-47 and 47-48 after 74,493 cycles. Panel failure occurred after 75,263 cycles when several multiple cracks linked together to form a single crack over 50 inches in length.

The cumulative frequency of cracking along the upper rivet row can be plotted as a function of minimum crack length and of total cycles. Figure 4 shows this type of plot for 3 different points in time during the life of the panel. Thus, after 50,000 cycles, 9% of the rivets in the upper row had cracks greater than 0.2 inch. Further, after 70,000 cycles (or when 93% of the panel fatigue life had been consumed), 13% of the rivets in the upper row had cracks greater than 0.4 inch, while 3% of these rivets were greater than 0.8 inch. The cumulative frequency of cracking is simply a function of cycles and load levels.

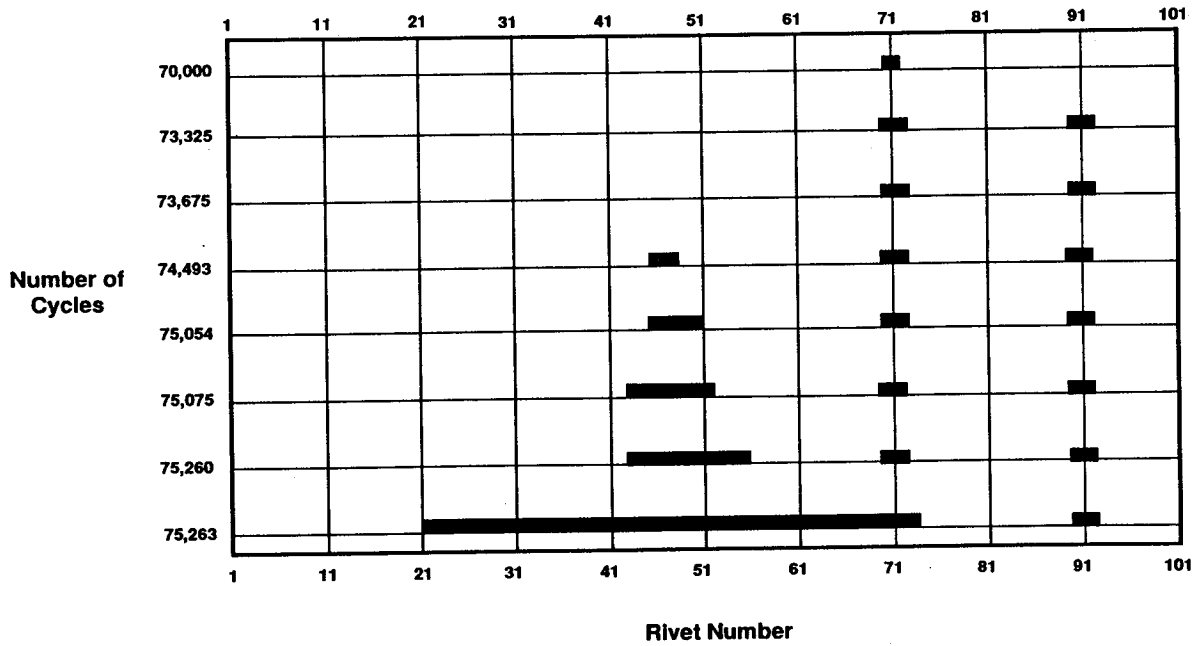


Figure 3. Sequence of multiple crack linkup events.

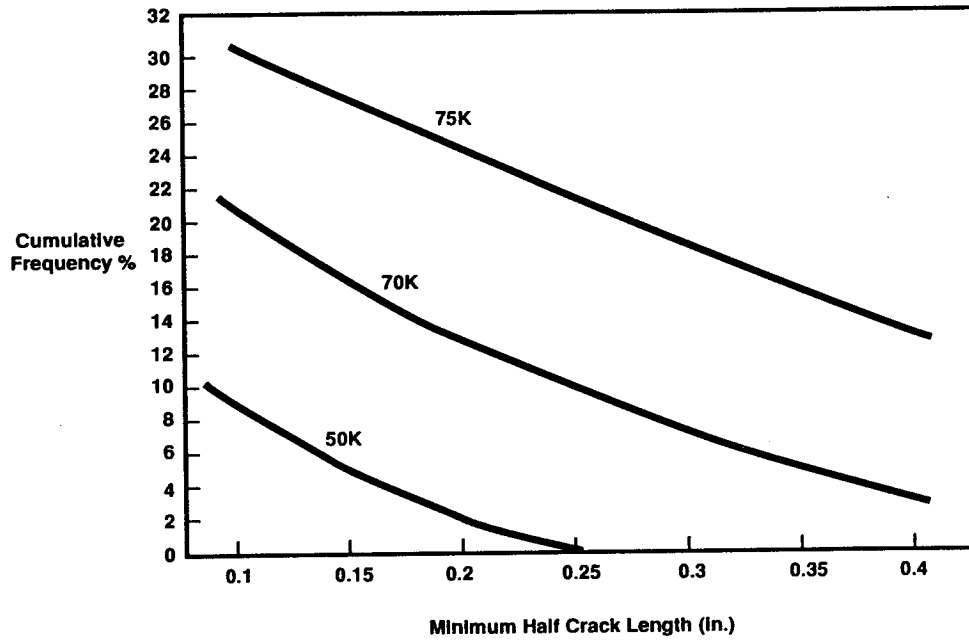


Figure 4. Cumulative frequency of cracking in baseline test.

TERMINATING ACTION EFFECTIVENESS TEST

Boeing Service Bulletin 737-53A 1039 Revision 3 [5] describes a remedial repair for fuselage lap splices. This repair has become widely known as the "terminating action" because its application terminates fatigue damage inspection. In 1987, the Federal Aviation Administration (FAA) issued Airworthiness Directive (AD) 87-21-08, mandating certain transport category aircraft to comply with this service bulletin. Basically, the repair entails replacement of shear head countersunk rivets in the upper row of the lap joint with universal head rivets which have a larger shank diameter.

The terminating action was applied to an initially undamaged test panel after 30,000 cycles, which is within specifications of the AD. Thus, all of the countersunk rivets in the upper row along the lap joint were drilled out and replaced with universal head rivets. Although three small cracks (all of which were less than 0.020 inch in length) were found at the time of the repair, they were completely drilled out when the holes were enlarged to accommodate the universal head rivets.

Some of the debonded tear straps were also repaired in this second panel, as further required by the Boeing Service Bulletin. All debonded tear straps in the regions two stringer bays above and one stringer bay below the lap joint were re-attached using universal head rivets. Specific details of the debonded tear strap repair are described in the Boeing Service Bulletin [5].

Cracking was observed in the *middle* row of rivets in the lap joint 37,800 cycles *after* the terminating action had been applied. Figure 5 shows a photograph of cracks in the middle rivet row of the panel with the terminating action repair. Middle rivet row cracking was also observed during fatigue tests conducted on flat lap splice coupons [4]. In Reference [4], the effect of different lap joint design parameters on the formation of multiple cracking was investigated in the laboratory. Middle row cracking was not observed, however, until after substantial cracking had occurred in the upper row. In addition, middle row cracking in a fuselage lap splice was observed in the Aloha airplane, as reported in Reference [1].

The application of the terminating action appears to retard the rate of crack growth. Crack growth curves for the baseline panel are compared to those for the terminating action repair panel in Figure 6. The rate of middle row crack growth *after* the terminating action is about one-third of upper row crack growth rate without the terminating action. Table 2 compares crack growth rates from other sources of data as well.

Figure 7 shows the distribution of cracks observed in the middle rivet row during cycling of the panel. (For purposes of comparison, the scale for crack length in Figure 7 is different than Figure 2.) Cracking appears to be concentrated in the midbay areas with relatively few cracks over the tear straps. This distribution is in contrast to the fairly random distribution observed in the baseline test.

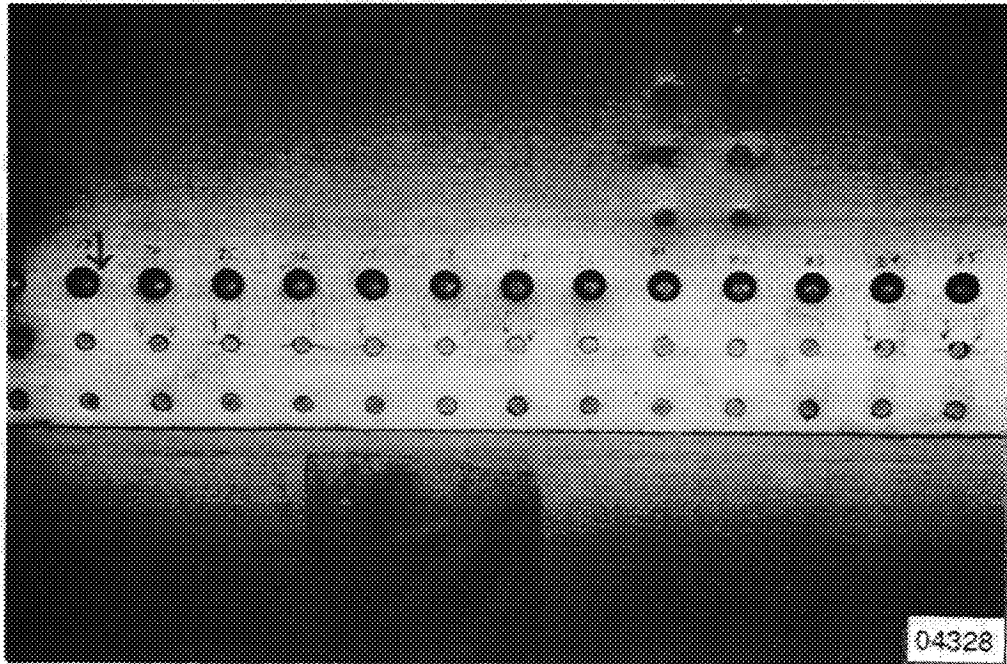


Figure 5. Middle rivet row cracking in terminating action repair test.

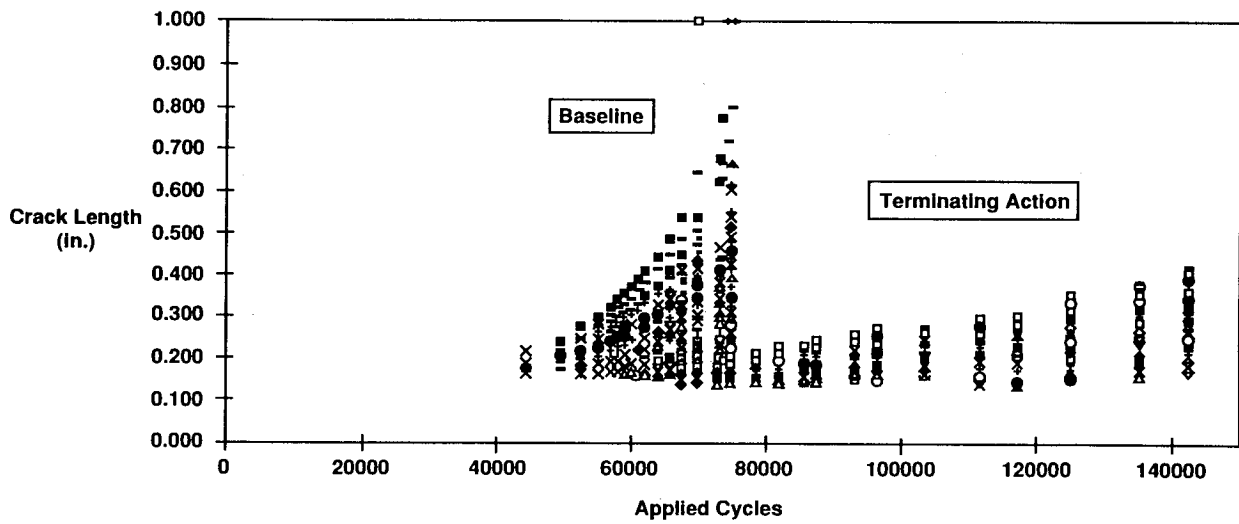


Figure 6. Crack growth rates observed during full-scale fatigue tests.

Table 2. Crack Propagation Rates from Different Sources

| Data Source | Skin Thickness (in.) | Hoop Stress (ksi) | Range of Crack Propagation Rate ($\mu\text{in./cycle}$) | | | | | | | | | | | |
|-------------------------|----------------------|-------------------|---|---|---|---|---|---|---|---|---|--|--|--|
| | | | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | | | |
| Baseline Test | 0.036 | 17.7 | | | | | | ▼ | — | — | — | | | |
| Aloha 243 | 0.036 | - | | | | | ▼ | — | — | — | | | | |
| FAA POD Panel 1 | 0.040 | 15.9 | | | | | | ▼ | — | — | — | | | |
| FAA POD Panel 2 | 0.040 | 15.9 | | | | | | ▼ | — | — | | | | |
| Terminating Action Test | 0.036 | 17.7 | ▼ | — | | | | | | | | | | |

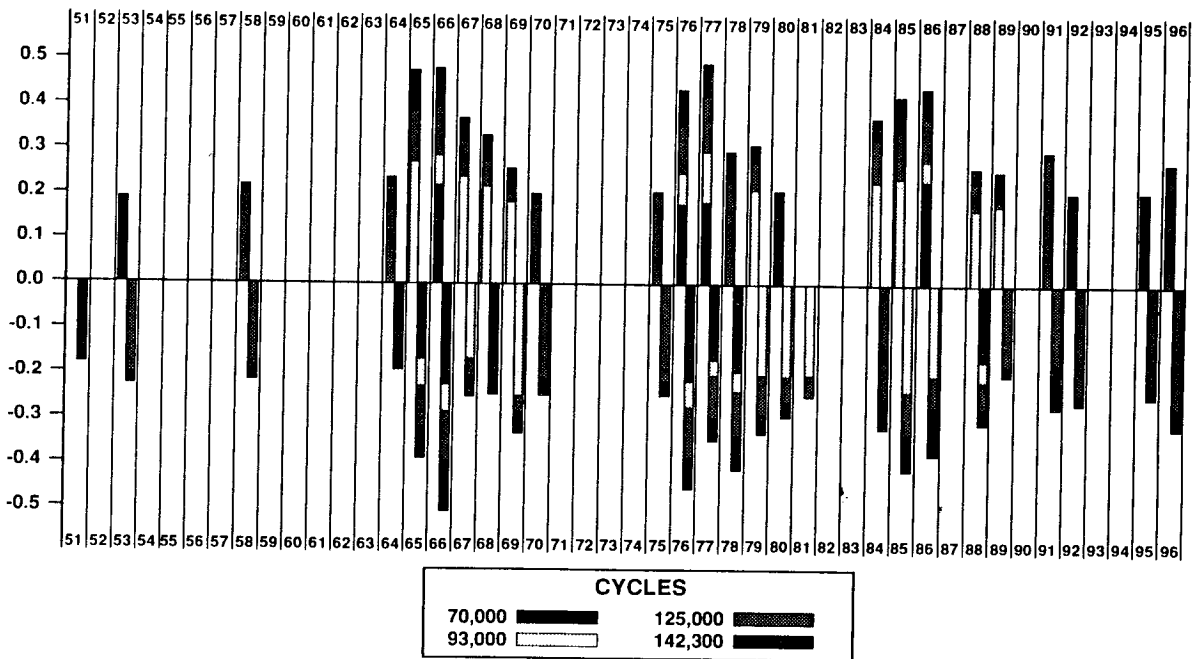


Figure 7. Distribution of cracks in middle rivet row in terminating action repair test.

Figure 8 shows the cumulative frequency of cracking in middle rivet row as a function of minimum crack length at 3 instances of time during the progression of the test. After 70,000 cycles, 17% of the rivets in the middle row had cracks greater than 0.2 inch in total length. At the termination of the test, 50% of the cracks had cracks greater than 0.30 inch.

The panel was fatigued for a total of 142,883 cycles. At that point, eighty-nine (89) cracks were found along the middle rivet row of the lap joint, but no linkup of cracks occurred. The longest of these cracks was measured to be 0.41 inch. The underside of the panel was inspected and no damage was found. Moreover, no cracks were detected in the upper row of rivets.

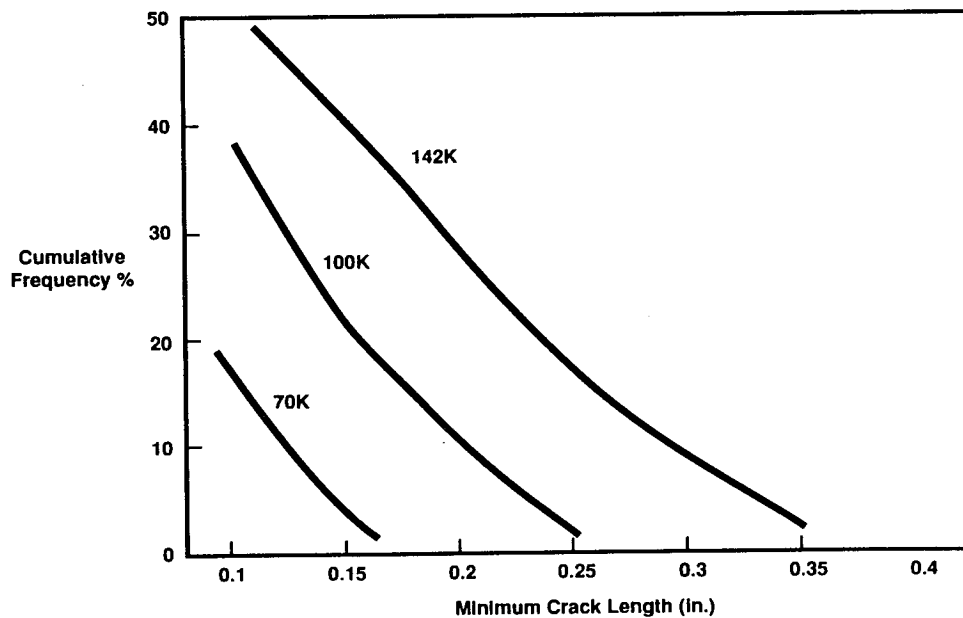


Figure 8. Cumulative frequency of cracking in terminating action repair test.

CONCLUSIONS

The results from two full-scale fatigue tests were reported in this paper. Based on the results from these tests, the following conclusions have been made.

- (1) Field representative WFD can be generated and quantitatively studied in the laboratory. The fatigue life of a lap splice panel loaded to a pressure differential of 8.5 psi is approximately 75,000 cycles. The first linkup of multiple cracks occurs at about 92% of this life.
- (2) The terminating action repair effectively eliminates cracking in the upper rivet row, and extends life by some margin.
- (3) Middle rivet row cracking could be expected at about 37,800 cycles *after* the terminating action had been applied to an initially undamaged panel. Although this middle row cracking in the fuselage panels with terminating action would not result in a catastrophic structural failure, it will draw the attention of maintenance staff and call for repair action which can be expensive.

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