EVOLUTION OF MULTIPLE-SITE DAMAGE IN THE RIVETED LAP JOINT OF A FUSELAGE PANEL

Abubaker Ahmed  
FAA-Drexel Fellowship Student  
FAA William J. Hughes Technical Center  
Atlantic City International Airport, NJ 08405, USA

John G. Bakuckas, Jr.  
Airworthiness Assurance Branch  
FAA William J. Hughes Technical Center  
Atlantic City International Airport, NJ 08405, USA

Jonathan Awerbuch, Alan C. Lau, and Tein-Min Tan  
Department of Mechanical Engineering and Mechanics  
Drexel University, Philadelphia, PA 19104, USA

Abstract
A full-scale fatigue test of a lap joint fuselage panel was completed as part of an experimental and analytical study of multiple-site damage (MSD) initiation and growth in a pristine structure. The panel was tested at the Full-Scale Aircraft Structural Test Evaluation and Research facility at the Federal Aviation Administration’s William J. Hughes Technical Center. A constant-amplitude load spectrum with underload marker cycles was used in the fatigue test to enable posttest fractographic examinations of the fracture surfaces. MSD cracks were detected emanating from rivet holes in the lap joint area. Crack linkup occurred in the outer skin layer along the upper rivet row in the lap joint. After linkup, crack growth rates increased substantially. Preliminary findings of the fractographic examinations indicate crack-tunneling behavior that can be attributed to several factors, including residual stresses resulting from the rivet clamp and bending of the lap joint. Data obtained from fractographic examinations correlated well with visual measurements made during the test.

Introduction
Multiple-site damage (MSD) is a source of widespread fatigue damage (WFD) that is characterized by the simultaneous presence of fatigue cracks in the same structural element [1]. Previous studies were conducted on fuselage panels at the Full-Scale Aircraft Structural Test Evaluation and Research (FASTER) facility at the Federal Aviation Administration’s (FAA) William J. Hughes Technical Center to examine the effects of multiple cracks on the fatigue crack growth and residual strength of large lead cracks. Panels with two different joint configurations, longitudinal lap joints and circumferential butt joints, were tested. Initial damage scenarios were inserted into the joints as either lead cracks only or lead cracks and multiple small cracks. Results from those studies showed that the presence of MSD reduced the number of cycles to grow the lead crack to the final length by approximately 37%. The load-carrying capacity of the panels was reduced by about 20% as a result of the presence of MSD [2].

The primary focus of the current study is on MSD initiation, growth, and interaction in a pristine fuselage panel subjected to fatigue loading. The test panel contained a longitudinal lap joint with four rows of countersunk rivets. A constant-amplitude fatigue loading simulating the fuselage cabin pressurization condition was applied using the FASTER test fixture. A quasi-static test was conducted prior to the fatigue test to ensure proper load introduction to the panel. MSD crack formation and linkup were monitored and recorded using the Self-Nulling Rotating Eddy-Current Probe system and the Remote-Controlled Crack Monitoring (RCCM) system. The fatigue test was terminated after observing the rapid growth of the lead crack, and the panel was then loaded quasi-statically up to final failure to measure the residual strength.
Fractographic examinations of the fracture surfaces in the panel are currently in progress to determine crack initiation sites and subsurface crack growth behavior. The fracture surfaces are being examined under the scanning electron microscope (SEM) to map marker band locations. Crack sizes and crack growth rates are measured and compared with the visual measurements to establish a complete crack growth history from initiation to failure. Preliminary findings of fractographic examinations confirm and correlate closely with visual observations of crack growth behavior made during the test.

**Test Facility**

As part of The National Aging Aircraft Research Program, the FASTER facility was established at the FAA William J. Hughes Technical Center for testing large curved panels representative of aircraft fuselage structures. The FASTER facility provides experimental data to validate and support analytical methods under development, including WFD prediction, repair analysis and design, and new aircraft design methodologies. The test fixture that is capable of simulating flight loading conditions encountered by an aircraft fuselage by applying the major modes of loading including internal pressurization; tensile hoop, longitudinal, and frame loads; and skin shear loads. Both quasi-static and long-term durability spectrum loadings can be applied in the FASTER facility. A detailed description of the FASTER facility and test fixture can be found in reference 2.

Two crack inspection methods, the Self-Nulling Rotating Eddy-Current Probe system and the RCCM system, were used to detect and monitor crack formation and propagation. The rotating-probe system is capable of detecting cracks that are hidden underneath countersunk rivet heads without having to remove the fastener. The system has a 15 mV output threshold; inspection signals exceeding the threshold indicate crack presence. Validation tests showed that the rotating-probe system has a 90% probability of detection for 0.032” electrical discharge machined (EDM) notch [3]. The RCCM system is a video data acquisition system consisting of two computer-controlled, high-precision x-y-z translation stages, each instrumented with a wide-field-of-view camera and a narrow-field-of-view camera. The combination of the two cameras allows monitoring the entire panel surface at several levels of magnification, providing a field of view ranging from 0.05” up to 14”. Each translation stage has a motion resolution of 0.00039” (1 µm), allowing accurate tracking of crack growth [2].

**Panel Configuration**

The pristine curved panel tested in this study was representative of a narrow-body fuselage structure fabricated per original equipment manufacturers specifications, Figure 1. The dimensions of the panel were 120” by 68” with a radius of 66”. The substructure of the panel included six frames extending in the circumferential direction with a 19” spacing and seven stringers in the longitudinal direction with a 7.5” spacing. The skin thickness was 0.063”. Both the skin and substructure were made of 2024-T3 aluminum. A longitudinal lap joint with two skin layers and two finger doubler layers was located in the middle of the panel along the middle stringer S4. The lap joint contained four rivet rows, labeled A, B, C, and D, respectively, as shown in Figure 1. The four edges of the panel were reinforced with aluminum doublers; holes were placed along the doublers so that the hoop and longitudinal load assemblies can be attached to the panel. The two ends of each frame, where the frame load assemblies are attached, were also reinforced with aluminum doublers. An elastomeric seal was bonded along the perimeter of the inner surface of the curved panel to attach to the pressure box of the FASTER fixture. The test panel was fully instrumented with strain gages to monitor and record strain distribution during the test. Strain gages were installed on the skin, frames, and stringers. Several back-to-back strain gage sets were installed at various locations on the skin to measure secondary bending.
sizes and number of cycles. This load spectrum, also known as a 6-4-10 spectrum, was applied in the corresponding number of marker bands on the fracture surface, thus enabling correlation between crack separated by spikes of ten 16-psi cycles. It is assumed that each group of marker cycles generates a follow-up with a group of marker cycles including either six, four, or ten iterations of 100 underload cycles consisted of three baseline blocks of 1000 maximum load cycles. Each of the baseline blocks was crack front shapes from posttest fractographic examinations. The load spectrum, shown in Figure 2, generate marker bands on the fracture surface to enable the reconstruction of crack growth history and maximum pressure was reduced to 12 psi during the underload cycles. Underload cycles were applied to Phase II, the fatigue test, was conducted using a constant-amplitude load spectrum including underload hoops and longitudinal directions, and tensile frame loads in the hoop direction, were applied to the panel. To accelerate the test, the maximum applied loads were higher than the typical operational loads of a fuselage structure. In phase I, prior to the fatigue loading, loads were applied quasi-statically to ensure proper load transfer from the fixture to the test panel. Strains were measured and recorded at all strain gages, and the strain distribution was studied to identify critical locations in the panel.

Table 1. Test phases and applied loads.

<table>
<thead>
<tr>
<th>Test Phase</th>
<th>Purpose</th>
<th>Load Type</th>
<th>Maximum Loads</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td></td>
<td></td>
<td>Pressure, psi</td>
</tr>
<tr>
<td>I</td>
<td>Strain Survey</td>
<td>Quasi-Static</td>
<td>16.0</td>
</tr>
<tr>
<td>II</td>
<td>Fatigue Crack Growth</td>
<td>Cyclic, R=0.1</td>
<td>16.0</td>
</tr>
<tr>
<td>III</td>
<td>Residual Strength</td>
<td>Quasi-Static</td>
<td></td>
</tr>
</tbody>
</table>

Phase II, the fatigue test, was conducted using a constant-amplitude load spectrum including underload cycles. A maximum pressure and a stress ratio of 16 psi and 0.1, respectively, were applied. The maximum pressure was reduced to 12 psi during the underload cycles. Underload cycles were applied to generate marker bands on the fracture surface to enable the reconstruction of crack growth history and crack front shapes from posttest fractographic examinations. The load spectrum, shown in Figure 2, consisted of three baseline blocks of 1000 maximum load cycles. Each of the baseline blocks was followed by a group of marker cycles including either six, four, or ten iterations of 100 underload cycles separated by spikes of ten 16-psi cycles. It is assumed that each group of marker cycles generates a corresponding number of marker bands on the fracture surface, thus enabling correlation between crack sizes and number of cycles. This load spectrum, also known as a 6-4-10 spectrum, was applied in the
FAA/USAF flat panel study [4] and was found effective in marking fracture surfaces in 2024-T3 aluminum.

![Figure 2. The marker band spectrum used in the fatigue test.](image)

The residual strength of the damaged panel was determined in phase III of the test by loading quasi-statically to failure. The internal pressure applied to the panel was increased in a 2-psi increment up to 14 psi, with a 1-psi increment thereafter. Strain gage measurements and crack lengths were recorded continuously throughout the residual strength test.

**Posttest Fractographic Examinations**

Comprehensive fractographic examinations of the crack surfaces in the test panel are underway to reconstruct crack growth histories. After the test was completed, the panel was removed from the test fixture and dissected to remove crack surfaces for examination under the SEM. Due to the final cracking configuration of the panel, the skin on the upper side of the outer rivet row was largely free from rivet clamp-up force, therefore, most crack surfaces in that rivet row were readily removable without having to break the fasteners. A rough cut was made using a cutting wheel to remove four bays of the free skin on the upper side of the crack. A Wilton 14-inch vertical band saw was then used to make precision cuts to remove individual crack surfaces.

The crack surface specimens were immersed in the citrus oil-based solvent d-Limonene/Ester for about 72 hours to soften the faying surface sealant so it could be removed without damaging the specimens. To remove corrosion products covering parts of the crack surfaces, the specimens were cleaned ultrasonically first in acetone, then in the nonchromated deoxidizer Turco Liquid Smut Go, and finally in distilled water. Cleaning time in the deoxidizer was about 3 to 5 minutes in a 1:6 deoxidizer to distilled water solution.

The fracture surfaces were examined using a FEI/Phillips XL30 Environmental SEM, compliments of the Department of Materials Science and Engineering at Drexel University, Figure 3. The XL30 is a computer-driven, high vacuum (10^-7 Torr) microscope with a field-emission electron gun and a spatial resolution of about 1.75 nm. The sample stage in the XL30 microscope is motorized for translation in the x, y, and z directions and rotation about the z axis. In addition, the sample can be manually tilted about the x axis. The imaging modes include secondary electron imaging, backscattered electron imaging, and mixed mode. Images can be stored digitally in a storage devise or transferred directly to a network location. Because of its high topographical contrast, secondary electron imaging was used for marker band detection and fracture surface imaging.

The following procedure described by Willard [5] was followed in mapping marker band locations. A local Cartesian x-y coordinate system was established on the fracture surface with the origin (0, 0) chosen at a reference point that could be readily identified under the SEM. Points along each marker band were then recorded relative to the origin. For each marker band, enough points were recorded to characterize its location and local curvature directions. All recorded points were then plotted in two-dimensional space, crack sizes and crack growth rates were measured directly from the plot. To minimize measurement errors, the plot was printed to a large scale.
Results and Discussion

Phase I, Strain Survey
Strain distributions in the panel were recorded under the quasi-static loading conditions described in Table 1. Results compared well with the results of a full-scale verification test conducted on an aft fuselage section of a narrow-body aircraft, thus verifying proper load transfer into the panel from the FASTER test fixture [6].

Back-to-back strain gage sets were installed at various locations on the skin to measure secondary bending. The schematic in Figure 4 shows four back-to-back gage sets located at a skin midbay (gage set 33), at a distance of 0.5" above rivet row A (gage set 38), midway between two rivets in row A (set 39), and 0.5" below rivet row D (gage set 45). The membrane strain component (the average of the strain measurements at the inner and outer surfaces) and bending strain component (half of the difference between the two measurements) for each gage set are shown in Figure 4. The membrane strain component is nearly the same at the four locations. The bending strain component, on the other hand, differs significantly; with the highest value being recorded at gage set 39 along rivet row A indicating severe bending. As a result of this severe local bending, the skin inner surface along the outer rivet row of the lap joint experienced high tensile stresses, making this rivet row a likely area for damage initiation.

Figure 3. The FEI/Phillips XL30 Environmental SEM.

Figure 4. Membrane and bending strain measured at two locations in the skin.
**Phase II, Fatigue Test**

During Phase II fatigue testing, crack formation in the lap joint was monitored and recorded using the Self-Nulling Rotating Eddy-Current NDI and high-magnification visual inspections using the RCCM. The first indications of damage formation in row A were high-level, eddy-current signals detected after 12,600 cycles, Figure 5. As shown in the figure, the number of rivets in row A with high-level signals subsequently increased. Few high-level signals were detected in Rows B and C. Rivet row D was not accessible for reliable inspection with the rotating probe system during the test because of the joint’s geometry.

![Figure 5](image_url) The percentage of rivet holes in lap joint with crack indications using eddy current.

The first visually detectable damage occurred after 51,500 cycles as a crack in the head of rivet A23 located in the outer row A between frames F2 and F3 (Figure 1). Similar rivet head cracks were detected at seven other rivets in row A. The crack growth from rivet A23 is illustrated in Figure 6 with a series of photographic images taken using the RCCM. The cycle number at which each image was taken is also shown in the figure. The crack in the rivet head grew along a curved path, which seemed to follow the perimeter of the rivet stem, Figure 6(a). Water leakage from the crack indicated that it was a through-thickness crack. It should be noted that the loading used in this study was much higher than what a fuselage would experience during normal service conditions. The rivets are not designed to sustain such high fatigue loads. Thus, it is believed that the rivet head crack initiated at the rivet shank-countersink interface due to the stress concentration in that area and propagated upwards to the surface. Apparently, the high local bending along rivet row A caused stress concentration at the root of the rivet head leading to crack initiation. After 80,550 cycles, a visually detectable skin crack developed on the right-hand side of the rivet at a distance from the edge of the rivet hole. The crack grew in both directions and eventually grew into the rivet hole, Figure 6(b) and (c). At a later stage, another through-thickness crack appeared on the left side of the rivet, Figure 6(d). This crack also grew in both directions and linked up with the rivet hole, Figure 6(e). Similar crack growth was observed at the neighboring rivet A22, Figure 6(f). Eventually, linkup occurred forming a large lead crack in the outer row, Figures 6(g) and (h).
All four cracks emanating from rivets A22 and A23 were detected by high eddy-current inspection signals before they were visually detected. The plot in Figure 7 shows the inspection history of rivet holes A22 and A23. High-level signals were first detected on the right side of A23, denoted A23-R, after 35,000 cycles, well before the crack was visually detected at 80,550 cycles. At both rivets, eddy-current inspections indicated damage presence on both sides after 70,000 cycles.

Figure 7. Change in maximum amplitudes of eddy-current signals recorded at rivet holes A22 and A23.
The crack growth from the first linkup of MSD between rivets A22 and A23 at 106,217 cycles to a two-bay crack after 107,458 cycles is shown schematically in Figure 8. The cycle number for each schematic is also shown in the figure.

Figure 8. Crack progression along rivet row A following the first linkup.
Shortly after the first linkup, Figure 8(a), MSD was visually detected emanating from rivet hole A24, Figures 8(b) and 8(c). Unlike the cracks at A22 and A23, the MSD cracks at A24 were completely linked to the rivet hole when first detected. On the right side, the lead crack and MSD crack from A24 grew towards each other quite rapidly and eventually bypassed each other forming a football shaped ligament prior to linkup, Figures 8(d) and 8(e). The lead crack continued to grow and eventually progressed into the neighboring rivet to the left, A21, Figure 8(f). No crack was detected at rivet hole A21 before the linkup. The lead crack then grew rapidly at both ends, Figures 8(g) through (k). On the right, the lead crack extended to the right through rivet hole A26 and linked up to one of the existing cracks at shear clip rivet F3-5. Shortly after that, a short crack was detected on the left side of rivet hole A27, Figure 8(l). By 107,458 cycles, the lead crack had grown slightly beyond rivet A27 to the right to about midway between rivets A16 and A17 to the left, Figures 8(m) and (n).

Figure 9 shows the growth of the lead crack as a function of fatigue cycles up to the end of fatigue test at 107,458 cycles. The left and right crack lengths are measured from the centerline midway between rivets A22 and A23 where the first linkup occurred. It can be seen that the lead crack initially grew at approximately the same rate in both directions. On the right side, as the crack-tip approached frame F3, the growth rate decreased significantly. However, growth of the left crack tip continuously increased and eventually grew noticeably with every cycle. At this stage, the fatigue test was terminated. By the end of fatigue test, the lead crack grew to a length of 16.04", extending along the outer rivet row of the lap joint mainly in bay two but crossing over frame F3 slightly into bay three.
Phase III, Residual Strength Test
The panel was then subjected to quasi-static pressurization up to failure to measure the residual strength. Loads were applied quasi-statically with a pressure increment of 2 psi up to 14 psi and an increment of 1 psi afterwards. Further crack extension was observed at 16-psi pressure. Catastrophic failure occurred at a pressure of 17.8 psi. The lead crack abruptly extended to a five-bay, 75” crack. Figure 10 shows the final state of damage after the residual strength test. The left crack tip of the lead crack extended to frame F2, linked up with the crack at shear clip rivet F2-5, and then stopped about 0.5” past rivet F2-5. The right crack tip extended much further, stopped between rivets A62 and A63 just before frame F6. As the crack extended through frames F4 and F5, it turned upwards and linked up with the cracks emanating from shear clip rivets F4-5 and F5-5 and skipped two or three rivet holes in row A before turning downwards to continue extending along the rivet row. The crack simply progressed unstably along rivet row A and displayed no crack turning or crack flapping capabilities. A posttest inspection on the inner surface of the panel showed that frames F3, F4, and F5 all fractured under the lap joint. The damage at frame F4 is shown in Figure 10.

Fractographic Examination Results
Crack initiation sites and subsurface crack growth behavior will be determined using fractographic examinations. Preliminary results are discussed here for a crack emanating from rivet A23 on the right side, designated A23-R. The SEM images in Figure 11 show a wide view of the crack surface and a close-up of a location along the faying surface between the outer skin and finger doubler. Fretting damage was evident and multiple crack origins indicate that the crack resulted from the coalescence of multiple small cracks from the faying surface. Crack origins were not observed along the bore surface of the rivet hole.

Tensile hoop strains measured on the inner surface of the outer skin from the high secondary bending of the skin along row A, shown in Figure 4, and the compressive residual stress field around the rivet hole from the rivet installation would contribute to crack initiation along the faying surface. Crack tunneling indicated more rapid growth in the longitudinal direction (along the faying surface) than in the transverse, through-the-thickness direction.
To determine subsurface crack growth behavior, beach marks inserted on the crack surfaces from the 6-4-10 marker band load spectrum shown in Figure 2 were used. Marker bands were readily detectable at magnification levels ranging from 350X to 2500X depending on the crack size. Marker bands become easier to detect and follow as the crack size increases. In general, the ten marker-band groups (10M) were easier to follow; the four marker band groups (4M) were a little more challenging. In the vicinity of the major crack origin, only 10M were detectable. Examples of SEM images showing the appearances of all three marker band groups, the 6M, 4M, and 10M are shown in Figure 12.

Figure 11. SEM images of crack A23-R showing: (a) a wide view of the fracture surface, (b) a close-up of crack origins locations along the faying surface.

Figure 12. SEM images of showing the appearance of: (a) a six marker band group, (b) a four marker band group, and (c) a ten marker band group.
The marker band locations were mapped throughout the area of the crack surface between the faying surface near the rivet hole and the location where the crack first penetrated the surface. A plot of all recorded marker band locations is shown in Figure 13. Crack size and crack growth rates were measured directly from this plot. As the plot shows, marker bands were detected back to the vicinity of the faying surface near the straight shank portion of the rivet hole. Near the crack initiation site, for very small cracks, only 10M were detected. No marker bands or striations were observed in the region next to the countersink surface of the hole, indicating the fatigue crack tunneled to the surface from the faying surface leaving the small ligament region, which eventually ruptured due to overload. As shown in the figure, crack front shapes were elliptical with the major axis along the faying surface and minor axis aligned in the through-the-thickness direction.

![Figure 13. Marker band locations for crack A23-R.](image)

The number of cycles associated with each marker band was based on the 6M marker band group closest to the surface where the crack was first visually detected at 80,500 cycles, as shown in Figure 13. Using this reference, the crack history was reconstructed and correlated with NDI and visual crack measurements made during the Phase II fatigue test. Measurements of the crack sizes and crack growth rates were made at three locations: (a) near the free surface, (b) along the geometric centerline of the specimen, and (c) along the faying surface, as illustrated in Figure 13.

During the Phase II fatigue test, this crack, A23-R, was first detected using eddy-current inspection after 35,000 cycles, Figure 7. From the marker band reconstruction of the crack history, the crack measured along the faying surface was approximately 0.05” after 35,000 cycles, Figure 13. As mentioned previously, the rotating probe system has a probability of detection of 90% for a 0.032” EDM notch. Complete correlation between eddy-current inspection signals and subsurface crack sizes will be made when more fractographic examination data is available.

Crack sizes obtained from fractographic examinations along the faying surface, the centerline, and near the free surface are compared with visual measurements in Figure 14. As seen, visual crack measurements correlate with the fractography results. Work is underway for longer cracks to compare the fractographic results with visual measurements.
Crack growth rates (da/dN) as a function of crack size are plotted in Figure 15. The plot shows growth rates in the longitudinal direction measured along the faying surface, the centerline, and near the free surface. Longitudinal crack growth rates are slightly higher near the free surface. Crack growth rates from visual measurements follow the general trend of those measured from fractography.
Concluding Remarks
Multiple-site damage (MSD) initiation, growth, and interaction in an initially undamaged curved fuselage panel containing a longitudinal lap joint were investigated. The test panel was subjected to a constant-amplitude fatigue loading using the Full-Scale Aircraft Structural Test Evaluation and Research facility located at the Federal Aviation Administration William J. Hughes Technical Center. Strain survey tests were conducted to ensure proper load introduction to the panel. During the fatigue test, damage formation and growth in the lap joint were monitored and recorded in real time using the Rotating Eddy-Current Probe system and high-magnification visual methods. Cracks were initially detected in the outer critical rivet row in the lap joint after 12,600 cycles using Rotating Eddy-Current Probe. Visually detectable cracks eventually developed in the rivet head of the critical rivet row after 51,500 cycles and then in the lap joint outer skin after 80,550 cycles. The first MSD linkup occurred after 106,217 cycles, forming a lead crack. Subsequently, the lead crack grew very rapidly along the outer rivet row eventually forming 16.04” two-bay crack after 107,458 cycles. A residual strength test was conducted with quasi-static loads to determine the load-carrying capacity of the panel. Catastrophic failure occurred at a pressure of 17.8 psi and the lead crack extended instantaneously across five bays to a final length of 75” without any crack turning or flapping. Three frames were fractured under the lap joint.

Fractographic examinations of the fracture surfaces were started on the first visually detected crack in the lap joint to determine initiation sites and mechanisms and to reconstruct subsurface crack growth behavior from marker band locations. Results from this single crack revealed multiple initiation sites and extensive fretting damage along the faying surface of the skin. A map of the locations of marker bands shows crack tunneling behavior before the crack penetrated the surface. The condition of the faying surface and the residual stresses from rivet clamp are two factors that affected subsurface crack growth behavior. Crack sizes and crack growth rates measured from fractography were correlated with eddy-current inspection results and with visual measurements made during the test. Fractographic examinations and correlations will be continued for this as well as the remaining cracks in the lap joint.

Acknowledgement
The first and the last three authors would like to express their sincere appreciation to the Federal Aviation Administration (FAA) William J. Hughes Technical Center for its support through the FAA-Drexel Fellowship Research Grant (97-G-032) to Drexel University.

REFERENCES
5. S. A. Willard; 1997; Use of Marker Band for Determination of Fatigue Crack Growth Rates and Crack Front Shapes in Pre-Corroded Coupons; NASA/CR-97-206291