AFRL-VA-WP-TR-2004-3109

DEVELOPMENT OF A MATHEMATICAL MODEL TO PREDICT CRACKING IN CORRODED AIRCRAFT STRUCTURES

Clare A. Paul

Analytical Structural Mechanics Branch (AFRL/VAO)
Structures Division
Air Vehicles Directorate, Air Force Materiel Command
Air Force Research Laboratory
Wright-Patterson AFB, OH 45433-7542

Gerhardus H. Koch

CC Technologies Laboratories, Inc.
2704 Sawbury Boulevard
Columbus, OH 43235

Le Yu and Noriko Katsube

The Ohio State University
Columbus, OH 43210

JUNE 1998

Final Report for 10 June 1997 – 10 June 1998

THIS IS A SMALL BUSINESS TECHNOLOGY TRANSFER PROGRAM (STTR) PHASE I REPORT.

Approved for public release; distribution is unlimited.

STINFO FINAL REPORT

AIR VEHICLES DIRECTORATE
AIR FORCE MATERIEL COMMAND
AIR FORCE RESEARCH LABORATORY
WRIGHT-PATTERSON AIR FORCE BASE, OH 45433-7542

NOTICE

This technical report has been reviewed and is approved for publication.

CLARE A. PAUL
Project Engineer
Analytical Structural Mechanics Branch
Structures Division

MAJ. JAMES W. ROGERS
Branch Chief
Analytical Structural Mechanics Branch
Structures Division

DAVID M. PRATT, Ph.D.
Technical Advisor
Structures Division

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YY)* June 1998

2. REPORT TYPE Final

3. DATES COVERED *(From - To)* 06/10/1997 – 06/10/1998

4. TITLE AND SUBTITLE
DEVELOPMENT OF A MATHEMATICAL MODEL TO PREDICT CRACKING IN CORRODED AIRCRAFT STRUCTURES

5a. CONTRACT NUMBER F33615-97-C-3215

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER 65502F

6. AUTHOR(S)
Clare A. Paul (AFRL/VAO)
Gerhardus H. Koch (CC Technologies Laboratories, Inc.)
Le Yu and Noriko Katsube (The Ohio State University)

5d. PROJECT NUMBER A04Z

5e. TASK NUMBER

5f. WORK UNIT NUMBER 0W

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Analytical Structural Mechanics Branch (AFRL/VAO)
Structures Division
Air Vehicles Directorate, Air Force Materiel Command
Air Force Research Laboratory
Wright-Patterson AFB, OH 45433-7542

CC Technologies Laboratories, Inc.
2704 Sawbury Boulevard
Columbus, OH 43235

The Ohio State University
Columbus, OH 43210

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Air Vehicles Directorate
Air Force Research Laboratory
Air Force Materiel Command
Wright-Patterson Air Force Base, OH 45433-7542

10. SPONSORING/MONITORING AGENCY ACRONYM(S) AFRL/VAO

11. SPONSORING/MONITORING AGENCY REPORT NUMBER(S) AFRL-VA-WP-TR-2004-3109

12. DISTRIBUTION/AVAILABILITY STATEMENT
Approved for public release; distribution is unlimited.

13. SUPPLEMENTARY NOTES
Report contains color. This is a Small Business Technology Transfer Program (STTR) Phase I report.

14. ABSTRACT
There is increasing concern about the possible detrimental effects of corrosion on the structural integrity of fuselage lap joints. Corrosion in lap joints can lead to a decrease in strength because of loss in skin thickness, early fatigue crack initiation caused by the formation of stress risers, and increased fatigue crack growth rates. The mode of corrosion in lap joints has generally been considered to be uniform loss of material. Based on the concept of general thickness loss and the formation of voluminous corrosion products as a result of exfoliation corrosion, models to predict the stress distribution and fatigue crack initiation sites have been previously developed. These models indicate that that the combination of loss in skin thickness and the build up of voluminous corrosion products inside the lap joint will lead to high stresses in the joint where fatigue cracks are likely to initiate. However, these calculations are not based on the actual morphology of corrosion in the lap joint.

15. SUBJECT TERMS
aging aircraft, corrosion, corrosion fatigue, environment, cracks

16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT:	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON (Monitor)
a. REPORT	b. ABSTRACT	c. THIS PAGE			Clare A. Paul
Unclassified	Unclassified	Unclassified	SAR	22	19b. TELEPHONE NUMBER *(Include Area Code)* (937) 904-8018

Standard Form 298 (Rev. 8-98)
Prescribed by ANSI Std. Z39-18

TABLE OF CONTENTS

LIST OF FIGURES

ABSTRACT

There is increasing concern about the possible detrimental effects of corrosion on the structural integrity of fuselage lap joints. Corrosion in lap joints can lead to a decrease in strength because of loss in skin thickness, early fatigue crack initiation caused by the formation of stress risers, and increased fatigue crack growth rates. The mode of corrosion in lap joints has generally been considered to be uniform loss of material. Based on the concept of general thickness loss and the formation of voluminous corrosion products as a result of exfoliation corrosion, models to predict the stress distribution and fatigue crack initiation sites have been previously developed. These models indicate that that the combination of loss in skin thickness and the build up of voluminous corrosion products inside the lap joint will lead to high stresses in the joint where fatigue cracks are likely to initiate. However, these calculations are not based on the actual morphology of corrosion in the lap joint.

In this paper, detailed metallography of a KC-135 lap-joint section describes the complex nature of corrosion on the contact or faying surface, with barely detectable corrosion penetrating deep into the skin. A finite element model was developed, based on the actual corrosion morphology of the lap joint. The finite element program ABAQUS was used to model the strain/stress distribution in a corroded lap-joint section. The corrosion was simulated by decreasing the skin thickness and applying a uniform pressure to represent the build up of corrosion by-products. A small hemisphere was introduced to simulate the localized intergranular corrosion. The results of the finite element analysis demonstrated that even a small hemispherical indent superimposed on uniform type corrosion near a fastener hole resulted in significant increase in elastic strain such that early fatigue crack initiation could be anticipated.

PREFACE

This work was performed under STTR Phase I Contract F33615-97-C-3215, which was administered under the technical direction of Mr. Clare A. Paul, AFRL/VASE.

1. INTRODUCTION

Over the past several years, much attention has been given to the phenomenon of multi-site damage (MSD), particularly of fuselage lap joints. The concern about MSD of lap joints was triggered by the infamous incident on April 28, 1988 where an Aloha Airlines Boeing 737 flying at 24,000 ft (7,300 m) suffered a structural failure in which an 18 ft (5.5 m) section of the fuselage crown was torn. The aircraft landed safely; however, one life was lost.

An accident/failure investigation by the National Transport and Safety Board (NTSB) concluded that the failure resulted from rapid and catastrophic crack growth caused by lap joint multiple site damage[1]. It is important to know, however, that this aircraft had operated its entire life in a tropical marine environment. Thus, severe corrosion due to exposure to this environment could have contributed to the formation of fatigue crack nucleation sites, which could have eventually lead to MSD and catastrophic failure. Extensive studies have been conducted to analyze and model MSD[2-5]. In these studies, MSD was generally considered to consist of small flaws, such as fatigue cracks originating from fastener holes, which could barely be detected by standard NDI techniques. Generally, these detectable cracks are greater than 0.05 inches. Again, environmental factors and corrosion must be taken into consideration. Although corrosion was considered to be a costly economic problem, it was not deemed to be an adverse factor in the structural integrity of critical airframe structures, such as fuselage lap joints[6].

Recently, much attention has been given to the phenomenon of pillowing of lap joints, where voluminous corrosion products at the contact or faying surfaces of a lap joint cause deformation of the skin, see Figure 1. Work by Komorowski and coworkers[7-9] indicated that the volume increase associated with the corrosion products is approximately 6.5 times the volume of the corroded parent aluminum alloy. The large volume increase and resulting deformation of the skin likely results in high stresses near the fasteners, which can have a definite detrimental effect on the structural integrity of the lap joint by lowering the strength and providing preferential sites for fatigue crack initiation. Mathematical and finite element modeling by Komorowski[7-9] were used to simulate the presence of the voluminous corrosion products within the fuselage lap joint. The stresses in the skin, which result from the internal pressure by the corrosion product build-up, from the fasteners, and from the reduction in thickness caused by material loss were all taken into account. The results of the calculations indicated that the pillowing significantly increases the stress in a fuselage lap joint, particularly in the vicinity of the fastener holes.

Moreover, fracture mechanics analyses conducted by Bellinger and Komorowski[9], and Welch[10] revealed troubling cracking behavior in lap joints subject to pillowing. The analyses suggested that cracks would form on the faying surface, forming a semi-elliptical crack front with a high aspect ratio without breaking through the outer surface of the skin. This would make visual inspection of the crack difficult. Also, analyses by Welch suggested that these cracks would propagate in directions other than perpendicular to the hoop stress.

2. APPROACH

2.1 Corrosion Characterization

While the above-discussed analyses were based on uniform loss of metal thickness as a result of crevice or exfoliation corrosion, no attempt was made to consider the actual lap joint corrosion morphology. It is reasonable to assume that localized corrosion, such as pitting or intergranular corrosion, could have an adverse effect on the stress distribution in the lap joint, and are a further detriment to the structural integrity of the joint.

A detailed characterization of the corrosion in a fuselage lap joint was performed using conventional metallographic techniques. A section of corroded fuselage lap joint was recovered from a KC-135 aircraft fuselage. This fuselage skin was made of aluminum alloy 2024- T3 sheet, and the original thickness of the lap joint section shown in Figure 2 was 1 mm (40 mils). After extensive visual inspection, areas for metallographic sectioning were selected. The selection of these areas was based on the appearance of corrosion both on the contact or faying surface and the outside surface. Generally, there was no evidence of corrosion on the outside surface of the skin, unless the corrosion had completely penetrated. The lap joint section was cut in different directions, longitudinally, transversely, and diagonally, so that the effect of orientation on the corrosion morphology could be examined.

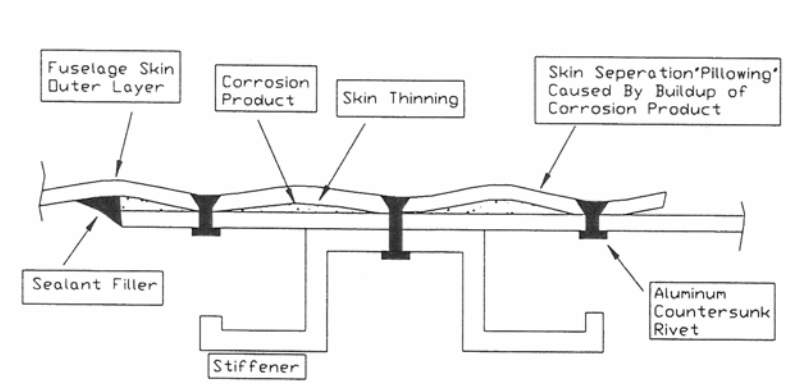

Figure 1. Cross Section through Lap Joint

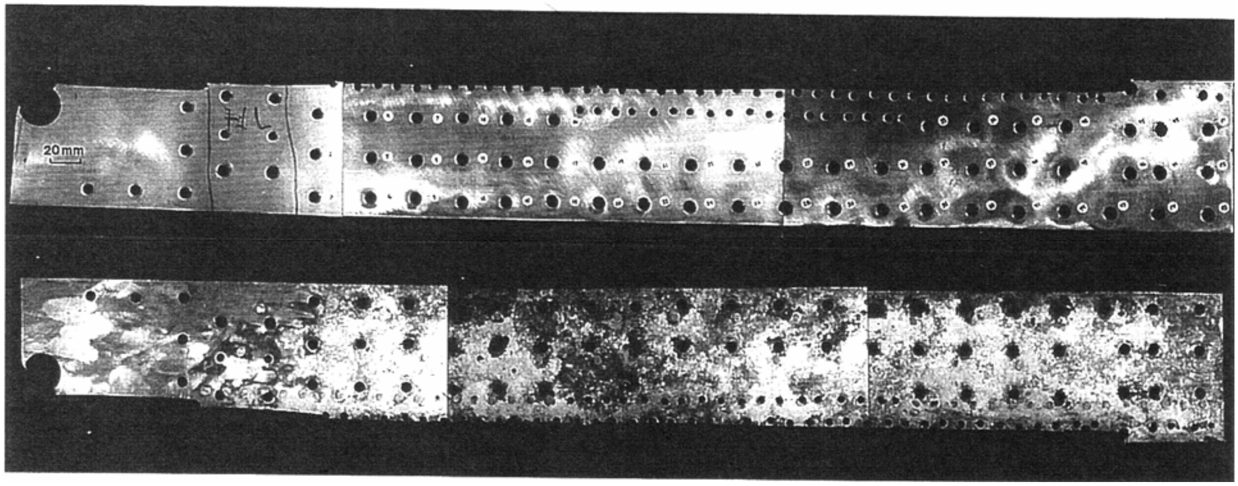

Figure 2. Contact (Top) and Outside (Bottom) Surface of KC-135 Fuselage Lap Joint

The cut sections were mounted in epoxy, which was cured under vacuum. The metallographic cross sections were ground and polished with colloidal silica and magnesium oxide to 2 µm finish. In order to be able to study the corrosion morphology in three dimensions, several parallel and perpendicular cross sections were made.

2.2 Finite Element Analysis

Finite element analyses (FEA) were carried out using ABAQUS and I-DEAS. The I-DEAS program has the capability to automatically generate both two- and three-dimensional meshes and is designed to interface with ABAQUS. In order to address the complex geometry of the type of corrosion in the fuselage lap joints, 10-node tetrahedron elements were used. The mesh of tetrahedron elements was automatically generated by I-DEAS and imported to ABAQUS for finite element calculation and post- processing.

The finite element analyses were performed on a 38 square mm (1.5 square inch), 10 mm (0.04 inch) thick sheet, which was reduced in thickness to 0.9 mm (0.036 inches) to simulate a 10 percent loss in thickness due to uniform corrosion attack. The distance between the four fastener holes in the coupon was 25.4 mm (1 inch), and the dimensions of the holes were modeled after those of the fuselage lap joint section examined in this study, tapered holes with a 3.5 mm (0.14 inch) inner radius and 4.5 mm (0.18 inch) outer radius. A hemispherical pocket representing a region of localized corrosion was superimposed onto the uniformly corroded area. The material properties used for the finite element analyses were those for aluminum alloy 2024-T3 with the Young's modulus 73,776 MPa of (10,700 ksi) and the Poisson's ratio of 0.33. Finally, linear elastic deformation was assumed in the analysis.

The strain distribution on the lap joint section were calculated, based on a 6.895×10^3 Pa (1 psi) uniform hydrostatic pressure, which simulates the pressure produced by the corrosion by-product. The 1 psi pressure was arbitrarily selected in order to simplify the calculations. Since elastic conditions are assumed, linear extrapolation to any value of pressure and resulting vertical displacement and strain can be carried out. Throughout the analysis, the symmetry condition was applied, so that one quarter of the boundary value problems could be analyzed. When necessary, the macro-micro analysis was used to improve the accuracy and efficiency of the model. Macro models based on a relatively coarse mesh were used first, and the calculated displacement fields were then used as boundary conditions for the fine-meshed micro models, which were used to represent the regions of localized corrosion.

3. RESULTS

3.1 Corrosion Characterization

As shown in Figure 2, the outside of the lap joint showed little or no evidence of corrosion, but on the contact or faying surface of the lap joint, voluminous corrosion products (hydrated Al_2O_3 or $Al(OH)_3$), which are characteristic of exfoliation corrosion, can be observed.

Metallographic cross sections through the lap joint section, such as in Figure 3, show extensive corrosion of the aluminum alloy skin, including pitting, exfoliation corrosion and intergranular corrosion. The figures show that exfoliation corrosion can either start from the faying surface or from a fastener hole. In the latter case, exfoliation can propagate over long distances before sufficient corrosion product is built up to create the pillowing effect. Once sufficient corrosion product has formed, lifting of the grains is observed, which is characteristic for severe exfoliation corrosion. When the grains are lifted off, and surrounded by the hydrated aluminum hydroxide, corrosion of the alloy grains continues by an apparent form of dealloying corrosion.

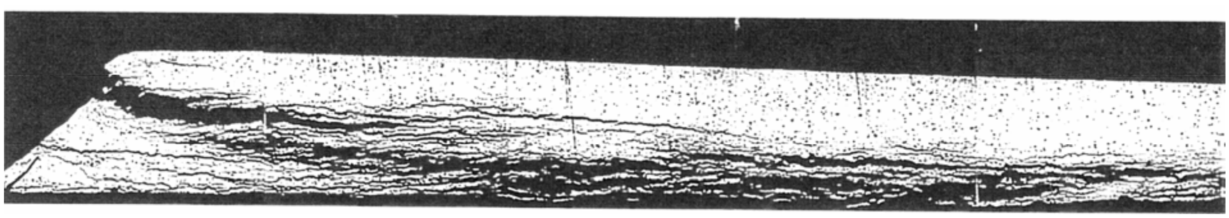

Figure 3. Metallographic Cross Section through Area Adjacent to Fastener Hole; Outside Surface is Facing Up.

The optical micrographs of exfoliated grains, shown in Figure 4, clearly demonstrate the dissolution of the grains. Specifically, the micrograph shows remnants of the corroded grains indicating selective dissolution.

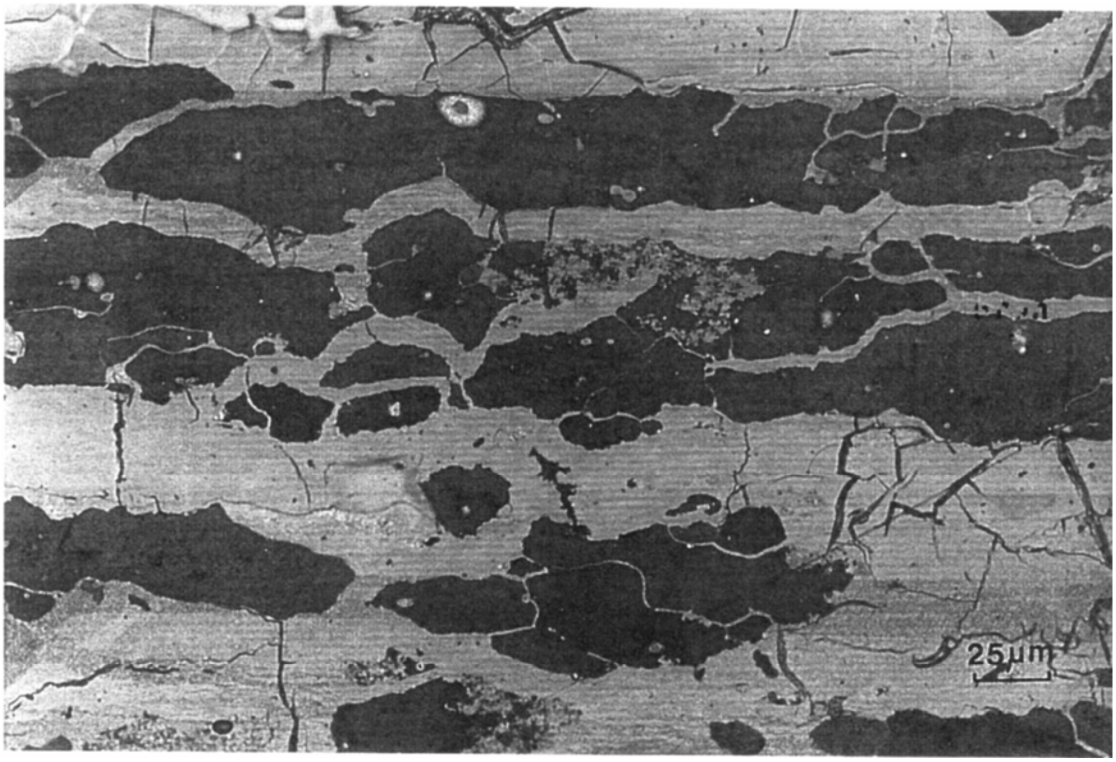

Figure 4. Metallographic Cross Section of Exfoliated Grain, Showing Preferential Dissolution of Aluminum from the Grains.

Further examination of several metallographic cross sections through the lap joint section demonstrated the presence of localized regions of intergranular corrosion in the aluminum alloy skin under the exfoliated regions. The intergranular attack, which occurs along grain boundaries and subgrain boundaries, is very tight suggesting that the voluminous aluminum hydroxide corrosion byproducts has not yet formed. Perpendicular cross sections such as shown in Figure 5 suggest that these regions of intergranular attack occur in the form of approximate hemispheres, and can propagate deep into the skin. In some cases, it was found that the intergranular attack has occurred along a single path, which is characteristic of intergranular stress-corrosion cracking.

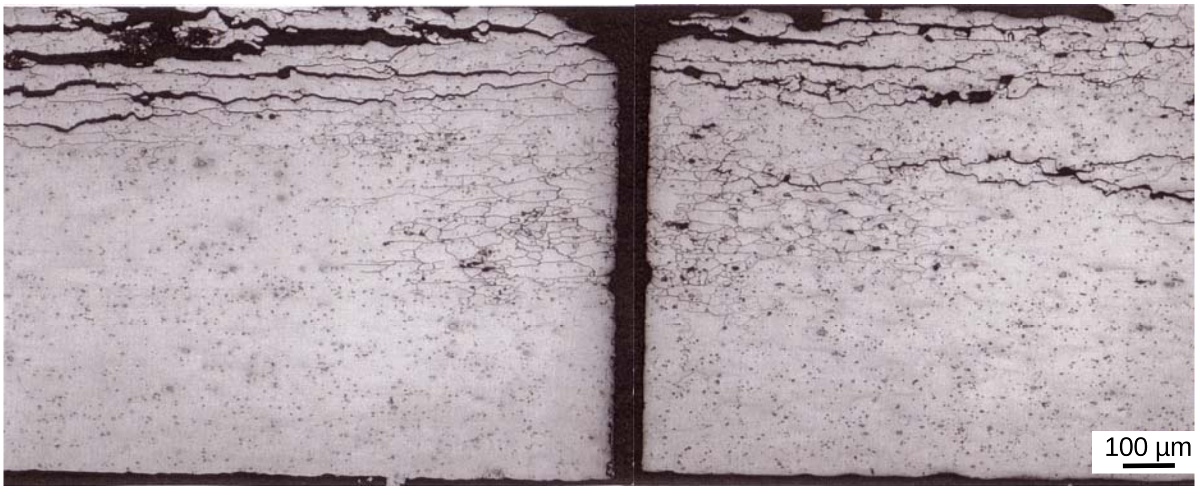

Figure 5. Optical Micrographs of Perpendicular Cross Sections Showing A Pocket of Fine Intergranular Corrosion.

3.2 Finite Element Analysis

The finite element modeling was conducted on a 38 square mm (1.5 square inch) plate section, 0.9 mm (0.036 inch) thick. The dimensions of the fastener holes, with an inner radius of 3.5 mm (0.14 inch) and outer radius of 4.5 mm (0.18 inch), were similar to those of the fastener holes in the fuselage lap joint. Because of the symmetry of the plate, one quarter of the plate was modeled using 3420 10-node tetrahedron elements. The displacement components of the fastener hole were assumed to be zero. When a uniform pressure of 1 psi was applied, the maximum normal strain in the diagonal direction of the plate was found to be 0.029 on the diagonal 5 mm (0.198 inches) from the center of the fastener hole.

In order to examine the effects of localized corrosion on the stress and strain distribution on the plate, a small hemisphere with a radius of 0.46 mm (0.018 inches) was placed on the diagonal 5 mm (0.198 inch) away from the center of the fastener hole. The location of the hemisphere was selected to coincide with the point where the normal strain in diagonal direction was the highest. With the 0.46 mm (0.018 inch) radius hemisphere, the maximum normal strain in diagonal direction is 0.0498, which occurs on the side where the hemisphere is located, see Figures 6 and 7. The strain at the location of the hemisphere is approximately 72% higher than the strain at the same location without the hemisphere. It is important to note that the calculations are based on the assumption of linear elasticity, and thus the magnitude of the maximum normal strain is well beyond the elastic range of the material.

After determining the effect of a 0.46 mm (0.018-inch) radius hemisphere on the strain distribution near a fastener hole, the effect of size of the hemisphere was investigated. When the 0.46 mm (0.018-inch) radius hemisphere was reduced by a factor of two, it was found that upon application of the 1 psi lateral pressure, the maximum normal strain was 0.053. This strain is 6.6% larger than the strain calculated for the larger hemisphere. The results of these calculations indicate that the size of the hemisphere has a significant effect on the stress-strain distribution around the hemisphere, and that there is likely to be an optimal size at which the strain is at a maximum.

Preliminary study indicates that a hemisphere with a radius of about a quarter of the plate thickness seems to cause the highest localized strain.

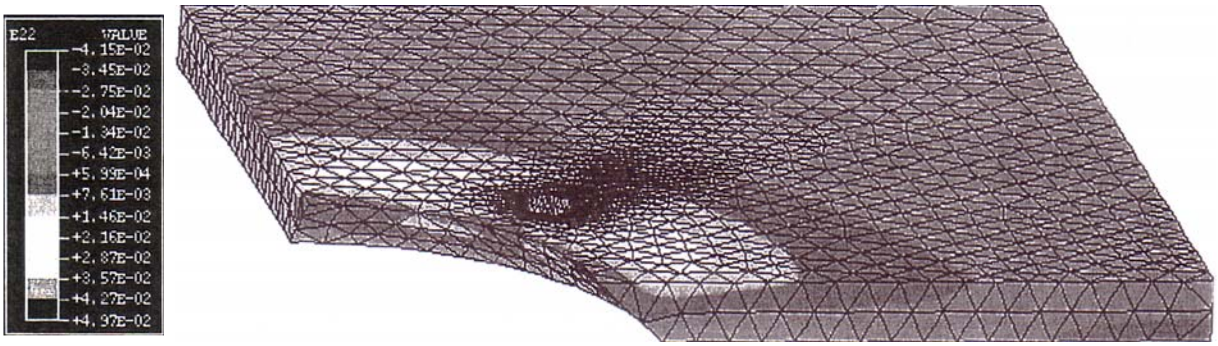

Figure 6. Finite Element Mesh of Quarter Sheet with Fastener Hole and Hemisphere with 0.46 mm (0.018 Inch) Radius and Located 5 mm (0.198 Inch) from Center of Fastener Hole.

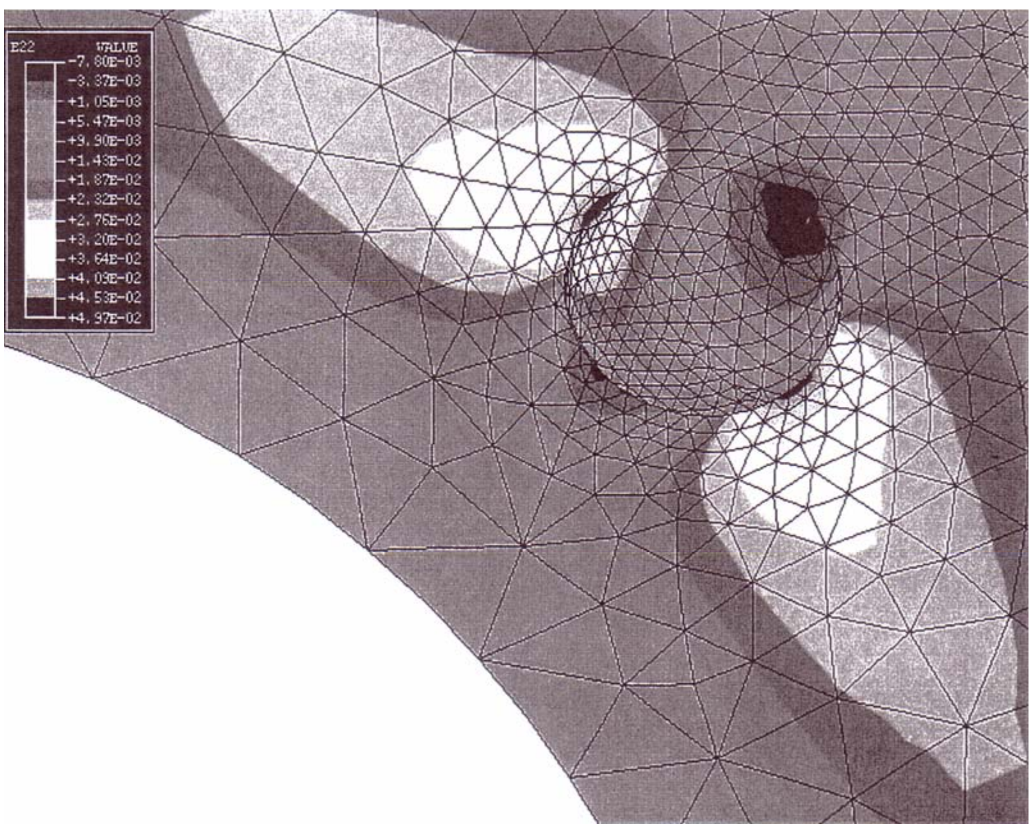

Figure 7. Local view of Strain Distribution Around the Hemisphere Shown in Figure 6.

3.3 Impact on Fatigue Life

Previous studies by Komorowski, et. al.[7-9] have shown that local stresses near the edge of a fastener hole due to faying surface corrosion can exceed the yield strength of aluminum alloy 2024. The cause for the increase in stress is corrosion product build-up and skin-thickness reduction.

Results of study by Perez, et al.[13] indicate that multiplying the stress concentration of a specimen with a hole ($K_t = 3.18$) by 1.21 can generally quantify the effect of chemical surface etching for Al 2124- T951. That is, the etching process causes pitting and the pit morphology essentially acts as a stress riser.

Therefore, it can be considered that corrosion on a faying surface provides at least three mechanical contributions to a reduction in fatigue life: (1) increase in local stresses around a fastener due to high-volume corrosion product build-up, and (2) the corrosion pit acting as a stress riser, and (3) the loss of load-bearing material acting to increase the local stresses. Subsequent analysis in this section will focus on the effect of the first two contributors to a reduction in fatigue life.

A first-order approximation of the reduction in fatigue life due to the increase in local stresses caused by the faying surface corrosion can be accomplished using strain-based fatigue analysis. The increase in local stresses caused by the build-up of corrosion product can be thought of as creating a constant local residual stress around the fastener hole. The increase in local stresses caused by the corrosion pit morphology (stress riser) may be accounted for by multiplying the stress concentration of the baseline geometry with a stress concentration associated with the corrosion pit morphology.

To that end, a strain-based fatigue life was performed using the parameter values shown below in Table 1.[11,12] S_{RES} represents the stress due to the faying surface corrosion product build-up given as a percentage of the yield stress and was accounted for by including it in Nueber's rule.

$$\sigma\varepsilon = \frac{\left(K_t S + S_{RES}\right)^2}{E} \qquad \text{Equation 1}$$

where K_t is the stress concentration, S is the remote stress, σ is the local stress, ε is the local strain, and E is the elastic modulus.

The increase in local stress due to the corrosion pit geometry (stress riser) was estimated.

$$K_t = K_t^{Hole} K_t^{Pit} \qquad \text{Equation 2}$$

where $K_t^{Hole} = 3.18$ is the stress concentration for an unfilled, uncorroded hole, and from Perez, et al.[13] $K_t^{Pit} = 1.21$ is an estimated equivalent stress concentration for corrosion pit morphology. Using the above values in Equation 2, a stress concentration, K_t , was calculated to be 3.85.

Table 1. Impact of Local Residual Stress, S_{RES}, on Strain-based Fatigue Life Prediction

		K_t = 3.18	K_t = 3.85	
S_{RES} (MPa)	S_{RES} (% S_Y)	*N*	*N*	Life Reduction Factor
0	0.00	4,383,381	860,198	5.1
30	0.10	2,740,545	601,614	4.6
61	0.20	1,816,462	445,208	4.1
91	0.30	1,273,926	348,024	3.7
121	0.40	945,788	285,539	3.3
152	0.50	741,709	243,555	3.0
182	0.60	610,061	213,964	2.9
212	0.70	521,159	192,145	2.7
242	0.80	458,155	175,414	2.6
273	0.90	411,469	162,157	2.5

S_{max} = 82.7 MPa (12 ksi)
Stress Ratio = 0.1

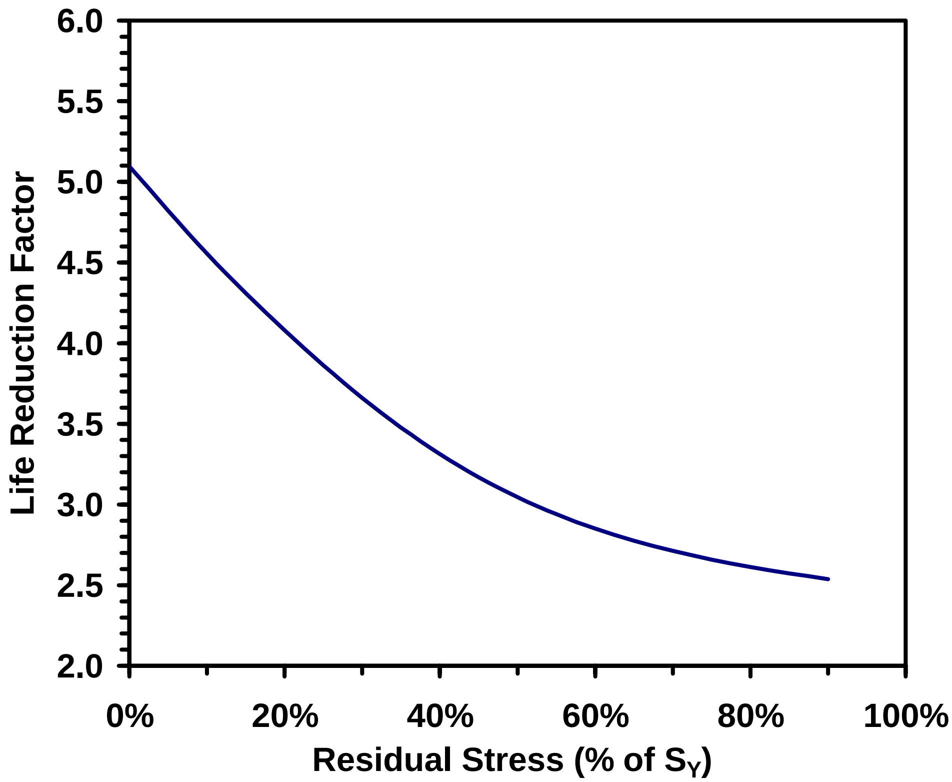

Figure 8. Reduction in Predicted Fatigue Life as a Function of Residual Stress

Not surprisingly, the strain-based fatigue life analysis predicts a significant reduction in fatigue life due to either "residual stresses" or "stress concentrations" resulting from corrosion. Combining the contribution from each yields a further reduction in fatigue life. And while the actual fatigue life predictions shown in Table 1 and Figure 8 may not represent the actual impact on USAF aircraft, the approach outlined may help form the underpinnings of a method to better model or assess the impact of corrosion on structural integrity.

4. DISCUSSION AND CONCLUSIONS

The results of this research strongly suggests that corrosion on the contact or faying surfaces of a fuselage lap joint can have a detrimental effect on the structural integrity of the joint. Contrary to the general assumption that corrosion in lap joints is uniform in nature, the metallographic results demonstrate that lap joint corrosion is complex and a combination of various forms of corrosion. The most obvious is exfoliation corrosion, which is a special form of intergranular corrosion common in the 2024- T3 aluminum alloy. Initially, the grain boundaries are attacked and hydrated aluminum oxide, $Al_2O_3.3H_2O$, or aluminum hydroxide, $Al(OH)_3$ corrosion byproducts form at the grain boundaries. Since the volume of the corrosion by products is approximately six times that of the aluminum it replaces, the surface of the skin tends to swell. Once the exfoliated grains are surrounded by the corrosion byproduct, the grains will continue to corrode by preferential dissolution of aluminum, forming aluminum hydroxide. When these voluminous aluminum hydroxide corrosion byproducts form in the confined space between the two contacting skins of a lap joint, considerable pressure is exerted on to the skins. Since in the case of the KC-135 fuselage lap joint, the inside skin of the lap joint is supported by a longeron, the pressure inside the joint will push out the outer skin. This will result in bulging between the fasteners, also known as pillowing.

It was demonstrated in the literature, and confirmed in the present FEA work that the pillowing resulting from the corrosion byproducts causes high stresses and strains in regions adjacent to the fastener holes. These regions could then become preferential sites for fatigue crack initiation. Moreover, it was found by Komorowski et al.[9] and Welch[10] that once cracks form, they develop semi-elliptical crack fronts with high aspect ratio's. These cracks do not break the surface and are therefore difficult to detect. Also, it was pointed out by Welch[10] that these cracks can propagate in directions other than perpendicular to the hoop stress, which could potentially lead to rapid loss of the lap joint function.

Detailed metallographic analysis of the lap joint section revealed that underneath the exfoliated grains regions of fine intergranular attack exist. These regions can be approximated by a hemisphere, which can deeply penetrate into the skin. Due to the tightness of the intergranular corrosion, it will be very difficult to detect and recognize these regions of corrosion with the common NDI techniques, until they have completely penetrated the skin. The hemispheric envelopes of the intergranular corrosion were simulated with FEA in order to determine their effects on the stress and strain distribution of a pillowing skin between four fasteners. The FEA indicated that under conditions of uniform loss of plate thickness, the maximum strain of the plate as a result of pillowing is already high near the fastener holes. When hemispheres are introduced at those locations of high strain, the resulting maximum strain increases significantly without affecting the vertical displacement of the plate. It was further found that the size of the hemisphere has a significant effect on the local maximum strain level. If the hemisphere is very small, its effect on the strain distribution is also small. If the hemisphere is relatively large (half the thickness of the plate), then the effect of uniform thinning and pillowing overwhelms the effect of the hemisphere. Thus, the localized high strain near the fastener hole depends on both the pillowing and the formation of areas of localized corrosion, and preliminary calculations indicate that a hemisphere with a radius of a quarter plate thickness results in the highest local strain.

The results presented lead to the conclusion that corrosion inside lap joints of aluminum alloy 2024- T3 is very complex. Some forms of corrosion observed are exfoliation corrosion, pitting, preferential dissolution, and intergranular corrosion. Due to the formation of voluminous corrosion byproducts inside the joint, the outer skin of the joint is pushed out, which results in high stresses around the fastener holes. The presence of local regions of intergranular corrosion under the exfoliated grains weakens the joint further by lowering the residual strength, and by creating possible sites for fatigue crack nucleation. Based on the FEA Results, stress analyses can be performed to account for the effect of corrosion fatigue life. While this type of analysis is useful in understanding the magnitude of the various contributing factors on the reduction of fatigue life, it will not be used to assess the life in U.S. Air Force aircraft. The U.S. Air Force requires the use of Damage Tolerance methodology to help assure the safe operation of its aircraft. The authors, and others, continue to develop corrosion metrics and transformations that will be used to assess the impact of corrosion on structural life using approaches based in fracture mechanics.

5. REFERENCES

1. "Aircraft Accident Report - Aloha Airlines, Flight 243, Boeing 737-200, N737111, near Maui, Hawaii, April 28, 1988", NTSB/AAR-89/03, National Transportation Safety Board Bureau of Accident Investigation, Washington, D.C., June 4, 1989.

2. Structural Integrity of Aging: Airplanes, Eds. S. N. Atluri, S. G. Sampath, and P. Tong, Springer-Verlag, New York, 1990.

3. Durability of Metal Aircraft Structures, Proc. of International Workshop on Structural Integrity of Aging Airplanes, March 31-April 2, 1992 Atlanta, Eds, S. N. Atluri, C.E. Harris, A. Hoggard, N. J. Miller, and S. G. Sampath.

4. FAA/NASA International Symposium on Advanced Structural Methods for Airframe Durability and Damage Tolerance, Hampton, V A, May 4-6,1994, Ed. C.E. Harris.

5. FAA/NASA Sixth International Conference on the Continued Airworthiness of Aircraft Structures, Atlantic City, NJ, June 27-28, 1995.

6. Lincoln, J. W. *The USAF Approach to Attaining Structural Integrity of Aging Aircraft.* Structural Integrity in Aging Aircraft, AD-Vol. 47, ASME 1995, pp. 9 - 19.

7. Bellinger N.C., S. Krishnakumar, and. P. Komorowski, *Modeling of Pillowing Due to Corrosion in Fuselage Lap Joints,* Canadian Aeronautics and Space Journal, Vol. 40, No.3, September 1994, pp. 125-130.

8. Bellinger, N.C., and J. P. Komorowski, and S. Krisnakumar, *Numerical Modeling of Pillowing Due to Corrosion in Fuselage Lap Joints,* LTR-ST-2005, National Research Council Canada, April 1995.

9. Bellinger, N. C., and J. P. Komorowski, *Implications of Corrosion Pillowing on the Structural Integrity of Fuselage Lap Joints,* Proceedings of the NASA-FAA Symposium on the Continues Airworthiness of Aircraft Structures, July 1997, pp. 391-401.

10. Welch, D. W., *Fracture Mechanics Analysis of a Corroded Aircraft Fuselage Lap Joint,* Master of Science Thesis, The University of Oklahoma, Norman OK, 1997.

11. Dowling, N. E., *Mechanical Behavior of Materials,* Prentice Hall, New Jersey, 1993

12. ASM Handbook, *Fatigue and Fracture* Vol 19, pp. 231, ASM International, Ohio, 1996

13. Perez, R., and C. Brook, et.. al., *Metrics Exploration and Transformation Development For Corrosion/Fatigue,* WL/FIBEC Control No. F33615-95-D-3216