AIAA JOURNAL
Vol. 41, No. 10, October 2003

Corrosion-Enhanced Fatigue and Multiple-Site Damage

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Multiple-site (fatigue) damage, (MSD) and its impact on the structural integrity (or safety of flight) of aging aircraft have been well recognized. Research to date has focused on fracture-mechanics-based analysis and experimentation for the consequences of MSD. The impact of corrosion on the early onset of MSD and the need for quantitative methodologies to predict the evolution and distribution of damage and MSD, on the other hand, are not fully appreciated. The mechanism for pitting corrosion in airframe aluminum alloys and the influence of pitting on the onset of fatigue cracking are briefly reviewed, and the influence of localized corrosion on the evolution of MSD is discussed. A mechanistically based probability model for corrosion and corrosion-enhanced fatigue crack growth and its application in predicting the probability of occurrence (PoO) of damage are summarized. The use of the PoO in a methodology to assess the onset and severity of MSD is demonstrated using teardown data from a Boeing 707 and two AT-38B aircraft.

Nomenclature

a	=	damage size, either pit depth or crack depth
a_o	=	initial pit radius
a_{tc}	=	depth of damage at the transition from a surface crack to a through-the-thickness crack
a_{tr}	=	depth of damage at the transition from a pit to a surface crack
b	=	simplification for $(2 - n_c)/2$
C_c	=	coefficient in the power law form for the crack growth rate equation
F	=	Faraday's constant
$F(x)$	=	cumulative distribution function (cdf); $F(x) = Pr\{X \leq x\}$
F_{tc}	=	geometric relationship for a through-the-thickness crack emanating from a circular hole
f	=	frequency
I_{Po}	=	preexponential constant in galvanic pitting current relationship
K_t	=	stress concentration factor due to a circular hole
M	=	molecular weight of the material
m	=	median used in the log-logistics cdf
N	=	number of cycles
n	=	valence
n_c	=	exponent in the power law form of the crack growth rate equation
$Pr(\ldots)$	=	probability
R	=	universal gas constant
r_o	=	hole radius
T	=	absolute temperature
t	=	time
t_{tc}	=	time of transition from a surface crack to a through-the-thickness crack
t_{tr}	=	time of transition from a pit to a surface crack
V	=	volume
ΔH	=	activation enthalpy for pitting
ΔK	=	driving force for fatigue crack growth
ΔK_{sc}	=	driving force for a surface crack
ΔK_{tc}	=	driving force for a through-the-thickness crack
$\Delta\sigma$	=	far-field stress range
μ	=	mean of a random variable
ρ	=	density of material

Received 3 June 2002; revision received 7 October 2002; accepted for publication 5 December 2002.

As we were putting final touches on this invited paper, we learned of the untimely passing of our friend John Lincoln. Jack devoted most of his professional career to aircraft structural technology and was the undisputed leader in the Air Force on structural integrity. His leadership, insight, and sage counsel will be sorely missed. We respectfully dedicate this contribution in his memory.

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Introduction

MULTIPLE-SITE (fatigue) damage (MSD) and its impact on the structural integrity (or safety of flight) of aging aircraft have been well recognized. Research to date has focused on fracture-mechanics-based analysis and experimentation of the consequences of MSD.[1] The impact of corrosion on the early onset of MSD and the need for quantitative methodologies to predict the evolution and distribution of damage and MSD, on the other hand, are not fully appreciated.[2] In this paper, the mechanism for pitting corrosion in airframe aluminum alloys and the influence of pitting on the onset of fatigue cracking are briefly reviewed. The influence of localized corrosion on the evolution of MSD is discussed. A mechanistically based probability model for corrosion and corrosion-enhanced fatigue crack growth and its application in predicting the probability of occurrence (PoO) of damage are summarized. The use and merit of the PoO in a methodology to assess the onset and severity of MSD are demonstrated through examples using teardown data from a Boeing 707 and two AT-38B aircraft.[3,4]

Pitting Corrosion and Fatigue Crack Growth

Gough[5] recognized the impact of pitting corrosion on fatigue cracking in the early 1900s. The mechanism of pitting corrosion and its impact on fatigue in aircraft aluminum alloys (such as 2024) are described in recent studies by Chen et al.,[6] Gao et al.,[7] Wei et al.,[8] Liao et al.,[9,10] and Liao and Wei.[11] These studies were motivated by the concerns with aging of commercial and military aircraft. They showed that pitting corrosion is induced by local dissolution of the matrix through its galvanic coupling with constituent particles in the alloys. These pits serve as nuclei for subsequent fatigue cracking and significantly reduce the serviceable life of a component or structure.[11] Plausible processes of aging, or damage accumulation, in airframe aluminum alloys therefore were considered to be dominated by localized (or pitting) corrosion in the early stage and by corrosion fatigue crack growth in the later stage as depicted in Fig. 1.

Simplified Mechanistically Based Probability Model

The essence of mechanistically based probability modeling for damage evolution and life prediction is the development of a time-dependent damage function that incorporates all critical internal

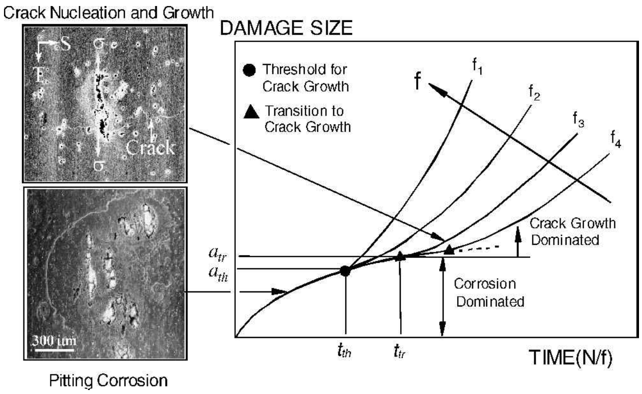

Fig. 1 Schematic diagram of the development of corrosion and corrosion fatigue damage.

(e.g., materials) and external (e.g., loading) variables and their variability. The damage function must integrate fracture and solid mechanics, electrochemistry and surface chemistry, materials science, and probability and statistics for damage nucleation and growth. Precise predictions of performance and risk for reliability analysis and life-cycle management are dependent on the accuracy of the damage function. One of the strengths of this approach is that it is iterative in that refinements including new information, insights, and data can be made. Further details and examples can be found in Harlow and Wei.[12] The approach is demonstrated herein.

The key random variable (rv) is the time-dependent damage size. Randomness in material properties and their sensitivity to environment are represented explicitly. Here damage is described by a single variable; i.e., the pit depth a during pitting and the crack depth or length a during fatigue cracking as described next.

Corrosion pits are assumed to be hemispherical with a constant volumetric growth rate, governed by Faraday's law augmented by a temperature-dependent Arrhenius relation. Pit depth a is given by Eq. (1):

$$a = \left\{[(3MI_{Po}/2\pi F\rho)\exp(-\Delta H/RT)]t + a_o^3\right\}^{\frac{1}{3}}, \quad \text{for} \quad a \le a_{\mathrm{tr}} \tag{1}$$

where a_o is the initial pit size (radius), a_{tr} is the transition pit size (or the size at crack nucleation), I_{Po} is the preexponential term in the Arrhenius relationship for the pitting current, $F = 96,514$ C/mol is Faraday's constant, $R = 8.314$ J/mol-K is the universal gas constant, and t is the time needed for a pit to develop to a depth of a. Values for the other parameters are for aluminum alloys, where $M = 27$ is the material molecular weight, $n = 3$ is the valence, $\rho = 2700$ kg/m^3 is the density, $\Delta H = 40$ kJ/mol is the activation enthalpy, and $T = 293$ K is taken as an average for the absolute temperatures when the aircraft is on the ground. Herein, a_o and I_{Po} are taken to be rvs. A power law model is used to represent the corrosion fatigue crack growth rate $(\mathrm{d}a/\mathrm{d}N)_c$, Eq. (2):

$$\left(\frac{\mathrm{d}a}{\mathrm{d}N}\right)_c = C_c \Delta K^{n_c} \tag{2}$$

The crack growth exponent n_c reflects the mechanistically deterministic dependence. The coefficient C_c is assumed to be a rv to reflect the variability in material properties with microstructure and environment. Also, the time-dependent number of loading cycles N is given by $N = ft$, where f is frequency.

The surface crack (sc) and through-the-thickness crack (tc) driving forces are given by Eq. (3):

$$\Delta K_{\mathrm{sc}} = (2.2/\pi)K_t \Delta\sigma\sqrt{\pi a}, \qquad \Delta K_{\mathrm{tc}} = F_{\mathrm{tc}}(a/r_o)\Delta\sigma\sqrt{\pi a} \tag{3}$$

where $\Delta\sigma$ is the far-field stress range, $2.2/\pi$ is for a semicircular crack in an infinite plate, $K_t = 2.8$ is the stress concentration factor for a circular hole, and $r_o = 3$ mm is the radius of the hole. Numerical values for $F_{\mathrm{tc}}(a/r_o)$ for an infinite plate under uniaxial tension containing a circular hole with a single through-the-thickness crack emanating from the hole perpendicular to the loading axis can be fitted empirically by Eq. (4):

$$F_{\mathrm{tc}}(a/r_o) = \{0.865/[(a/r_o) + 0.324]\} + 0.681 \tag{4}$$

Let t_{tr} and t_{tc} be the time at which a pit transitions into a surface crack and the time at which the surface crack transitions into a through-the-thickness crack, respectively. When $t_{\mathrm{tr}} \le t < t_{\mathrm{tc}}$, the driving force is ΔK_{sc}, and a is found to be given by Eq. (5) for $n_c > 2$:

$$a = \left[a_{\mathrm{tr}}^b + bfC_c\left(2.2K_t\Delta\sigma\big/\sqrt{\pi}\right)^{n_c}(t - t_{\mathrm{tr}})\right]^{1/b}$$
$$b = (2 - n_c)/2 \tag{5}$$

When $t \ge t_{\mathrm{tc}}$, a is obtained implicitly by using ΔK_{tc} from the following relationship:

$$t = t_{\mathrm{tc}} + \frac{1}{fC_c\left(\Delta\sigma\sqrt{\pi}\right)^{n_c}}\int_{a_{\mathrm{tc}}}^{a}\frac{\mathrm{d}a}{\left[F_{\mathrm{tc}}(a/r_o)\sqrt{a}\right]^{n_c}} \tag{6}$$

where a_{tc} is the size of the damage at t_{tc}. Numerical integration is needed for Eq. (6).

Corrosion fatigue crack nucleation reflects competition between pitting and crack growth. Criteria for transition have been proposed and validated in Chen et al.[13]:

$$\Delta K \ge \Delta K_{\mathrm{th}}, \qquad \left(\frac{\mathrm{d}a}{\mathrm{d}t}\right)_{\mathrm{crack}} \ge \left(\frac{\mathrm{d}a}{\mathrm{d}t}\right)_{\mathrm{pit}} \tag{7}$$

where ΔK_{th} is the threshold driving force, assumed to be a rv, and the derivatives are the appropriate time-based corrosion fatigue crack and pit growth rates.

Statistical variability is modeled through I_{Po}, a_o, C_c, and ΔK_{th}, which are chosen to be mechanistically and statistically independent of time. Scatter in material properties, environmental sensitivity, and resistance to fatigue crack growth is reflected in C_c. Material and manufacturing quality is depicted by a_o and ΔK_{th}. Finally, I_{Po} reflects the scatter associated with the electrochemical reaction for pit growth. The three-parameter Weibull cumulative distribution function has been found to characterize the rvs; see Harlow and Wei.[14]

Evidence for In-Service Development of MSD

Teardown data from the wing skins of a 24-year-old commercial Boeing 707 aircraft serve to illustrate the processes of damage evolution and to provide evidence of corrosion-enhanced evolution of MSD. The data also serve to demonstrate the efficacy of the mechanistically based probability modeling approach. The observations were made as a part of the U.S. Air Force Joint Surveillance, Target and Attack Radar System (J-STARS) program[6] to convert retired Boeing 707 aircraft for this service. The Boeing 707-321B (s/n 19266, line 531; designated as CZ-184), in service from delivery on 30 November 1966 to termination on 2 October 1990, had 57,382 flight hours and 22,533 flight cycles. This aircraft had the highest time for the 300 series aircraft in the inventory. For the inspected sections, a total of 494 multiple hole-wall cracks (MHWCs) were reported in Hug[3] for the lower wing skins. Approximately 5800 holes were inspected of which 578 (or about 10%) had some type of observable damage. Of the 5800, steel fasteners were used in 3300 holes along stiffeners S-4, S-5, S-7, S-8, and S-14, where 478 (or about 14%) were found to have damage. In comparison, the percentage of damaged holes without steel fasteners in them was less than 4%. Specifically, of the 578 damaged holes, 392 had MHWCs, of which 290 were reported to have MHWCs in one of the two highly stressed regions of the hole, and the remaining 102 had MHWCs in both (for a total of 494 MHWCs). Figure 2 shows the typical localized corrosion and corrosion-induced multiple fatigue cracking in a fastener hole. The distribution of damage for the largest MHWC in each fastener hole is shown in Fig. 3. The teardown report[3] should be consulted for details and additional observations.

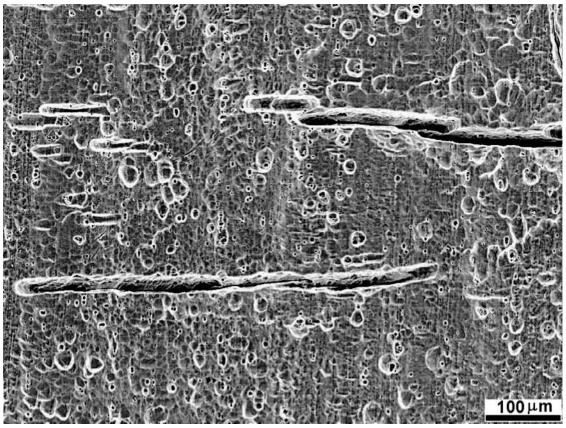

Fig. 2 Scanning electron micrograph showing localized corrosion and corrsion-related fatigue damage on the fastener hole wall of the CZ-184 aircraft. (Vertical direction of micrograph corresponds to the loading direction.)

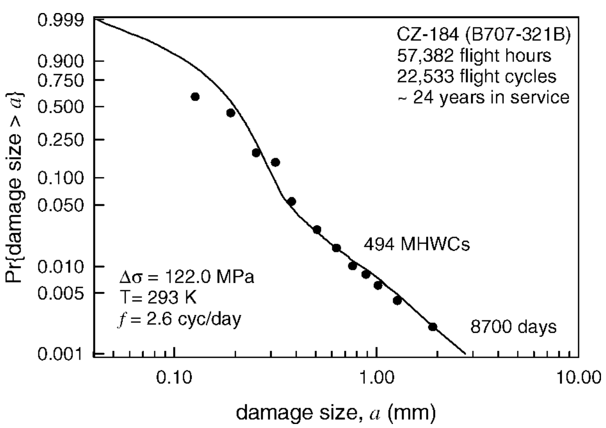

Fig. 3 Distribution of the largest reported MHWC in each fastener hole from the CZ-184 aircraft.[2,3]

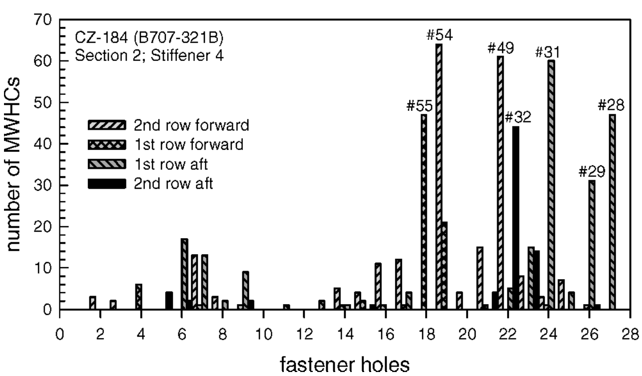

Fig. 4 Density of measured MHWC in 110 fastener holes in the lower wing panels of the CZ-184 aircraft.[2,3]

To perform a statistical analysis for a significant number of fastener holes, 110 contiguous holes along S-4 were investigated.[2] The fraction of holes with damage in this set along S-4 is quite high, with 51 of the 110 holes showing damage. A possible contributing factor to this high incidence is that steel fasteners were used along S-4. Galvanic coupling between the fasteners and the 2024-T3 aluminum alloy skin, in the presence of a deleterious environment, would enhance the nucleation and growth of localized corrosion. Figure 4 shows the density of the MHWCs for these 110 holes. The holes are numbered arbitrarily left to right, and each of the four rows is numbered separately. There are three holes with at least 60 cracks, and another three with at least 40. Furthermore, all six of these holes are close to each other.

Prediction of Damage Evolution, Distribution, and MSD

Damage distributions determined from teardown inspections for two Boeing 707 (designated as CZ-180 and CZ-184)[3] and two AT-38B[4] (designated as SP-0260 and SP-0283) aircraft are shown in Fig. 5. The difference in the severity of damage between the aircraft types reflects, likely, differences in operational profile, utilization (approximately 1200 vs 300 flights per year), and time on the ground. The evolution and distribution of corrosion and corrosion fatigue damage are predicted, through Monte Carlo simulation, using the simplified model described. Laboratory pitting corrosion and fatigue crack growth data on the appropriate alloys and nominal ground–air–ground bending loads (adjusted for nominal gust or maneuver loading) were used.

The effectiveness of this approach is demonstrated by the good agreement between the predicted distribution and that of the observed damage in the lower wing panels (2024-T3 aluminum alloy) of the CZ-184 aircraft (Fig. 3). More detailed measurements of damage,[2,15] based on the 110 contiguous fastener holes, showed that much of the smaller damage was not resolved during the initial teardown inspection.[3] The distribution of damage after 22,533 flight cycles is replotted in Fig. 6, along with the predicted distribution (solid curve) from the simplified model, after a small adjustment in the pitting current. (Note that the observed and predicted damage distributions were nearly identical for the largest damage in the fasteners hole and the damage within a given fastener hole.[15]) Using the same data, the evolution of damage was estimated for flight cycles from 15,000 to 35,000 and is also shown in Fig. 6.

Unlike the CZ-184 aircraft, there is no detailed experimental determination of corrosion damage on the AT-38B aircraft,

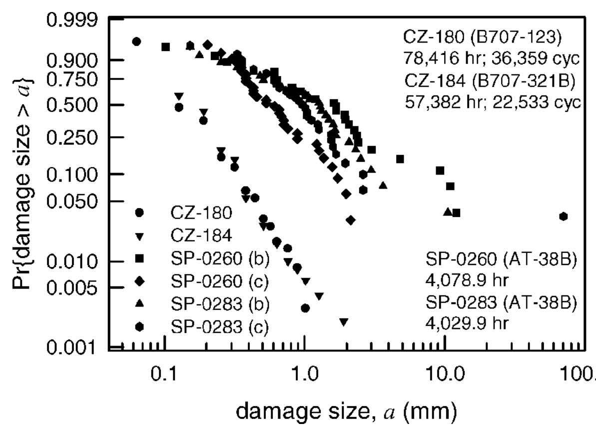

Fig. 5 Comparison between damage distributions in Boeing 707 and AT-38B aircraft.[2–4] (For AT-38B, b and c designate bore and corner cracks, respectively.)

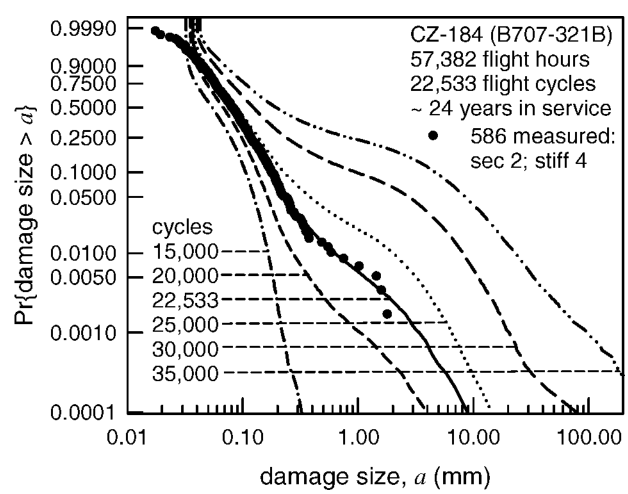

Fig. 6 Evolution and distribution of corrosion and fatigue damage in the CZ-184 aircraft.[2,3]

although evidence for corrosion was noted in the teardown inspection report.[4] Nevertheless, the mechanistically based probability model described herein is used to estimate the distribution of damage in the lower wing panels, made of 7075-T6 aluminum alloy. To conform to the Boeing 707 analysis, only the hole-wall cracks are considered, and the estimated distributions are shown in Fig. 7. The agreement is quite good, even though the influence of variable amplitude loading was not considered in this simplified estimation; departure at the larger damage sizes most likely reflects the simplification in the assumed loading. The significant impact of corrosion is inferred from a comparison between the estimated distributions, with and without corrosion, with the estimated damage sizes dramatically lower in the absence of corrosion.

On the basis of the estimated evolution and distributions of damage, the evolution of the spatial distribution of damage may be estimated through Monte Carlo simulation. Here the simulation is illustrated through the use of the data from the CZ-184 aircraft, and the progressive increase in the size of damage from 20,000 to 35,000 flight cycles is shown in Fig. 8. (Note that the aircraft had been retired after 22,533 flight cycles.) The simulation was carried out for 1000 fastener holes, and the abscissa in Fig. 8 refers to sequential numbering of sides of the holes. By using the same starting seed numbers, each of the successive simulations represents the same set of 1000 fastener holes and the locations of damage are retained. The rapid increase in the size of damage and development of areas

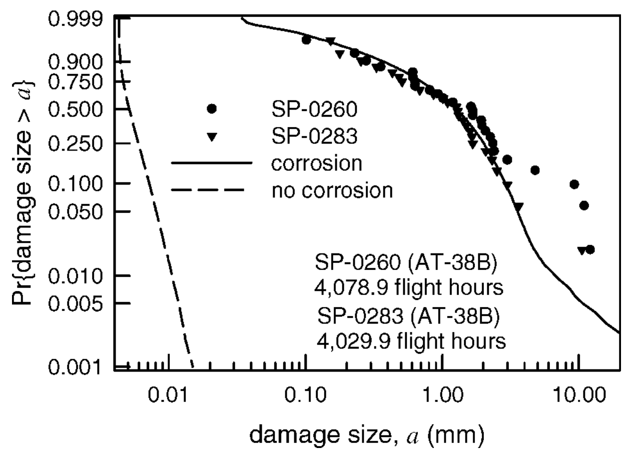

Fig. 7 Damage distribution in AT-38B aircraft.[2,4] Solid and dashed lines represent predicted distributions with and without corrosion, respectively.

of significant MSD following 25,000 flight cycles may be seen in Fig. 8. It should be noted that, for these simulations, the loading and external environmental conditions were taken to be constant over the 1000 fastener holes. The spatial variations in damage sizes reflect only variability in the rvs in the mechanistically based models. Spatial and time variations in external conditions may be imposed to better reflect actual operating conditions.

In practice, successive simulations will be made using a new set of randomly generated seed numbers to select the rvs in the models for each simulation. The results of each simulation (a realization) would represent what could happen at successive 1000 fastener holes in the same aircraft or at the same 1000 fastener hole location of different aircraft in a fleet, assuming that the external conditions remain the same. Figures 9 and 10 represent two such realizations showing the projected distribution in corrosion and corrosion fatigue damage over 1000 fastener holes following 30,000 flight cycles. They show the differences in damage severity and in the 25-hole areas of significant MSD. (The selection of the number of fastener holes to be used in the simulations is somewhat arbitrary. A total of 1000 holes is used here to ensure fidelity and to attempt to capture some of the more extreme occurrences.)

Following the same procedure, a spatial distribution of corrosion and corrosion fatigue damage over 1000 fastener holes in an AT-38B aircraft following 5500 flight cycles was simulated, using the estimated distribution shown in Fig. 7. The sizes of distributed damage are shown in Fig. 11 along with three areas of significant MSD. Comparisons of the results for the AT-38B aircraft (Fig. 11) with those for the CZ-184 aircraft (Figs. 8–10) show the impact of the more severe flight loads on the AT-38B aircraft and, implicitly, a greater influence of corrosion associated with the longer time that these aircraft had spent on the ground.

Discussion

These results further demonstrate the efficacy and utility of a mechanistically based probability approach for predicting the onset and evolution of distributed damage that can lead to areas of significant MSD. The early onset of fatigue damage and MSD is enhanced by localized corrosion and corrosion-enhanced fatigue crack growth. Distribution in damage is directly related to the randomness of key chemical and microstructural variables and is captured through appropriate mechanistically based probability models. It can be affected by local variations in loading and environmental conditions and inappropriate choices of materials in design. The very large damage, shown in Figs. 8–11, is in excess of the interhole spacing of about 25 mm and is beyond the range of validity of the simplified crack growth model. The degree of agreement with

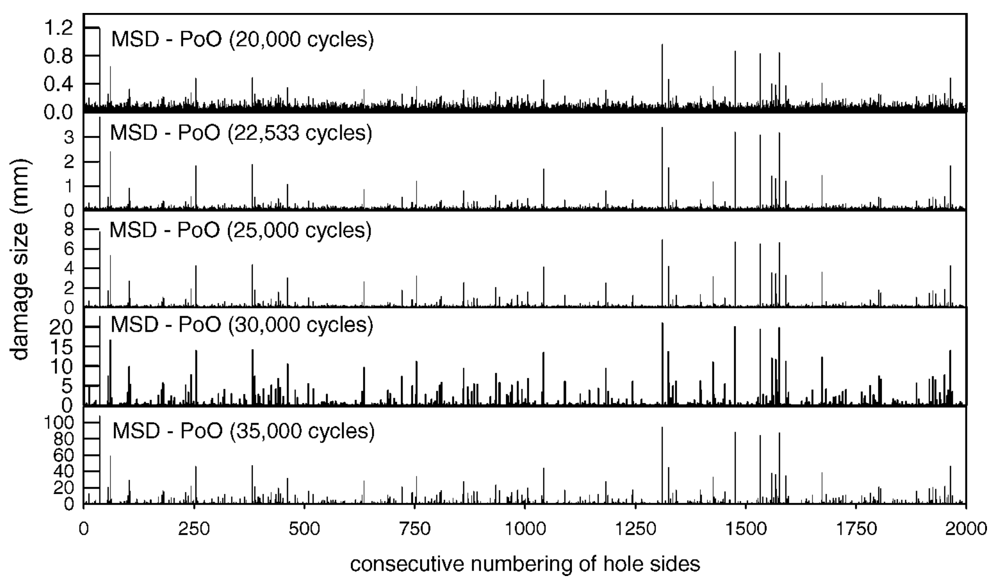

Fig. 8 Successive simulation showing the evolution and distribution of corrosion and corrosion fatigue damage and the formation of significant areas of MSD over 1000 fastener holes for the CZ-184 aircraft.

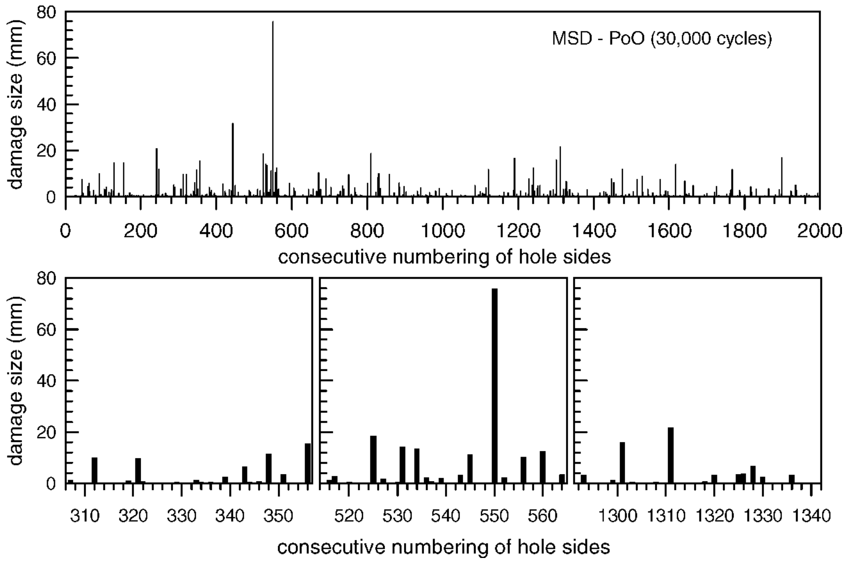

Fig. 9 One simulation showing the projected distribution of corrosion and corrosion fatigue damage over 1000 fastener holes and three 25-hole areas of MSD for the CZ-184 aircraft after 30,000 flight cycles.

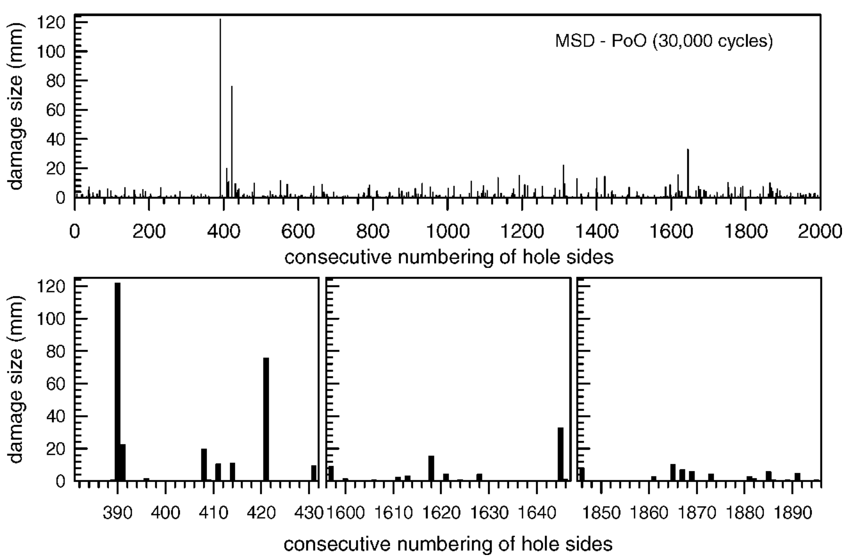

Fig. 10 Second simulation, showing the projected distribution of corrosion and corrosion fatigue damage over 1000 fastener holes and three 25-hole areas of MSD for the CZ-184 aircraft after 30,000 flight cycles.

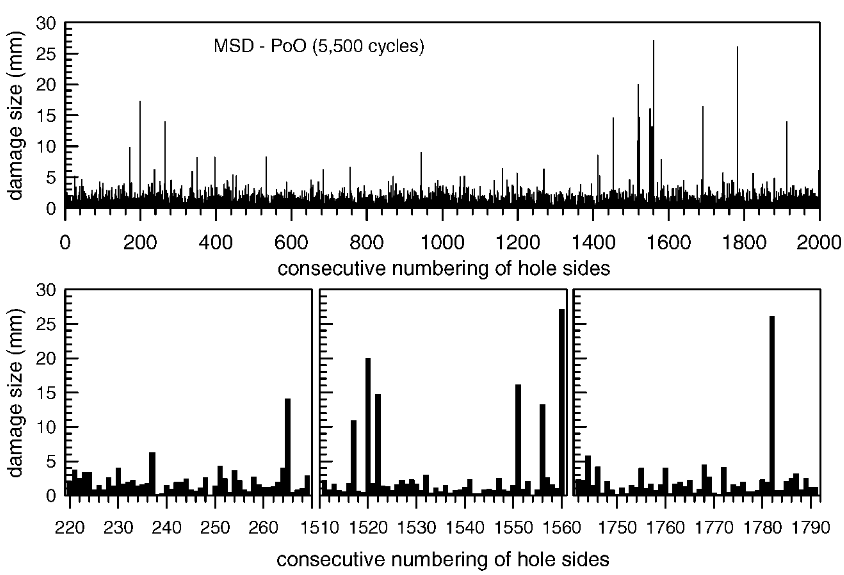

Fig. 11 Simulation showing the distribution of corrosion and corrosion fatigue damage over 1000 fastener holes and three 25-hole areas of MSD for the AT-38B aircraft after an estimated 5500 flight cycles.

teardown data, though, is encouraging and argues for the development and validation of refined models for use in design and fleet management and sustainment. Because early damage on the order of tens and hundreds of micrometers (e.g., in the form of corrosion pits) can significantly reduce fatigue lives[2] and cannot be readily detected by current nondestructive inspection techniques, there is an urgent need for the development of validated methodologies, such as the one described herein, for use in aircraft design and fleet management and sustainment.

Summary

A simplified, mechanistically based probability model for corrosion and corrosion-enhanced fatigue crack growth and its application in predicting the probability of occurrence (PoO) and spatial distribution of damage are summarized. The use and merit of the PoO in a methodology to assess the onset and severity of MSD is demonstrated through examples using teardown data from a Boeing 707 and two AT-38B aircraft. The onset of fatigue and MSD is a natural consequence of the randomness in the chemical and microstructural characteristics of the material and may be impacted by local variations in loading and environmental conditions and material selection. The significant impact of corrosion is shown. The value of the approach and the need for developing improved models for aircraft design and fleet management and sustainment are indicated.

Acknowledgments

This research is supported in part by the Air Force Office of Research under Grant F49620-98-1-0198 and Alcoa.

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