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Residual Strength of Stiffened LY12CZ Aluminum Alloy Panels with Widespread Fatigue Damage

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Abstract

Experimental and analytical investigations on the residual strength of the stiffened LY12CZ aluminum alloy panels with widespread fatigue damage (WFD) are conducted. Nine stiffened LY12CZ aluminum alloy panels with three different types of damage are tested for residual strength. Each specimen is pre-cracked at rivet holes by saw cuts and subjected to a monotonically increasing tensile load until failure is occurred and the failure load is recorded. The stress intensity factors at the tips of the lead crack and the adjacent WFD cracks of the stiffened aluminum alloy panels are calculated by compounding approach and finite element method (FEM) respectively. The residual strength of the stiffened panels with WFD is evaluated by the engineering method with plastic zone linkup criterion and the FEM with apparent fracture toughness criterion respectively. The predicted residual strength agrees well with the experiment results. It indicates that in engineering practice these methods can be used for residual strength evaluation with the acceptable accuracy. It can be seen from this research that WFD can significantly reduce the residual strength and the critical crack length of the stiffened panels with WFD. The effect of WFD crack length on residual strength is also studied.

Keywords: stiffened panel; widespread fatigue damage (WFD); residual strength; stress intensity factor; plastic zone linkup criterion

1 Introduction

An aging aircraft accumulates fatigue damage in the form of small scale cracks at the places of high stress concentration. This type of damage is commonly referred to as widespread fatigue damage (WFD). The two types of WFD are multiple site damage (MSD) and multiple element damage (MED)[1]. The residual strength of an aircraft structure will be significantly reduced by the existence of the small cracks adjacent to the rivet holes and much lower than that of only a single lead crack being concerned without considering the interaction with the surrounding cracks [2].

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The economic factors have impelled the commercial and military aircrafts to be used beyond their original design life. And the WFD of these aircrafts is an issue of great concern. Data have shown that for the aging aircrafts the WFD may reduce the residual strength for about 25% comparing with the case of only a lead crack being involved[3]. Considerable efforts have been devoted to studying the WFD behavior of aging aircraft panels and its effects on aircraft structure integrity. Several approaches have been used to estimate the residual strength of aircraft with WFD[4-7]. Swift[8] developed an analytical method to determine the residual strength of a panel based on the yield stress with taking into account the plasticity and crack interaction effects. Several other techniques have also been proposed for establishing stress intensity factors of WFD cracks, for example the finite element method (FEM)[9].

In this paper, the residual strength of stiffened LY12CZ aluminum alloy panels with WFD is calculated by both analytical method and FEM. The effects of WFD on the residual strength of the stiffened panels are studied. The tests of the stiffened panels with WFD are conducted to confirm these analyses.

2 Plastic Zone Linkup Criterion

Residual strength is defined as the maximum load a panel can bear before complete failure occurs. In comparison with the case of only a lead crack being involved, it is very difficult to evaluate the residual strength of the stiffened panel with WFD because of the interaction of the cracks. To this day, different residual strength criteria have been presented for the WFD[10].The plastic zone linkup criterion proposed by Swift is used here for its convenience and applicability[8].

The plastic zone linkup criterion assumes that when the plastic zone at the lead crack tip linkup with the plastic zones of the adjacent WFD crack as shown in Fig.1, the structure will fail. This can be expressed by

$$R_{\text{lead}} + R_{\text{msd}} = b \tag{1}$$

where R_{lead} is the size of lead crack plastic zone, R_{msd} is the size of multiple crack plastic zones, and b is the distance between lead crack tip and the adjacent WFD crack tip.

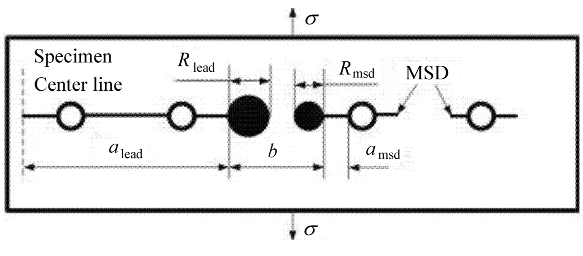

Fig.1 Schema of plastic zone linking up with multiple cracks.

With the Dugdale model, the plastic zone radius at crack tip under plane stress state can be expressed as

$$R = \frac{\pi}{8}\left(K/\sigma_{\text{ys}}\right)^2 \tag{2}$$

where K is the stress intensity factor at the crack tip; σ_{ys} is the yield strength of the material. When the plastic zone radius of Dugdale expression is adapted, the residual strength of stiffened panels with WFD can be expressed as

$$\sigma = \frac{\sigma_{\text{ys}}}{\pi}\sqrt{\frac{8b}{a_{\text{lead}}\beta_{\text{lead}}^2 + a_{\text{msd}}\beta_{\text{msd}}^2}} \tag{3}$$

where σ is the predicted residual strength of stiffened panel, a_{lead} is the half of the lead crack length, a_{msd} is the half of the adjacent multiple crack length, β_{lead} is the correction coefficient of the stress intensity factor of lead crack, and β_{msd} is the correction coefficient of the stress intensity factor of adjacent multiple crack. β_{lead} and β_{msd} can be obtained through Eqs.(4)-(5) respectively:

$$K_{\text{lead}} = \sigma\sqrt{\pi a_{\text{lead}}}\beta_{\text{lead}} \tag{4}$$

$$K_{\text{msd}} = \sigma\sqrt{\pi a_{\text{msd}}}\beta_{\text{msd}} \tag{5}$$

where K_{lead} and K_{msd} are the stress intensity factors of the lead crack and the adjacent multiple crack tips respectively. The stress intensity factors at crack tips can be calculated with engineering compounding method[11]:

$$K = K_0 + \sum_{n=1}^{N}\left(K_n - K_0\right) \tag{6}$$

where K_0 is the basic stress strength factor, K_n is the stress intensity factor for the nth simple boundary case, N is the total number of these simple boundary case.

3 Finite Element Analysis

The K-apparent criterion is used to determine the residual strength of the stiffened panels[12].A simulated wing structure panel of transport aircraft is chosen for this analysis; the original " I " cross section stiffeners are considered as straps riveted on the stiffened panel. The structure and size are illustrated in Fig.2. The rivets is made of LY10.4×14