International Journal of Fatigue 33 (2011) 323–331

Contents lists available at ScienceDirect

International Journal of Fatigue

journal homepage: www.elsevier.com/locate/ijfatigue

The lead crack fatigue lifing framework

L. Molent [a,*], S.A. Barter [a], R.J.H. Wanhill [b]

[a] Air Vehicles Division, Defence Science and Technology Organisation, Australia
[b] Aerospace Vehicles Division, National Aerospace Laboratory NLR, The Netherlands

ARTICLE INFO

Article history:
Received 17 May 2010
Received in revised form 27 August 2010
Accepted 13 September 2010
Available online 18 September 2010

Keywords:
Fatigue crack growth
Fatigue modelling
Life prediction
Fatigue testing
Fractography

ABSTRACT

A fatigue lifing framework using a lead crack concept has been developed by the DSTO for metallic primary airframe components. The framework is based on years of detailed inspection and analysis of fatigue cracks in many specimens and airframe components, and is an important additional tool for determining aircraft component fatigue lives in the Royal Australian Air Force (RAAF) fleet. Like the original Damage Tolerance (DT) concept developed by the United States Air Force (USAF), this framework assumes that fatigue cracking begins as soon as an aircraft enters service. However, there are major and fundamental differences. Instead of assuming initial crack sizes and deriving early crack growth behaviour from back-extrapolation of growth data for long cracks, the DSTO framework uses data for real cracks growing from small discontinuities inherent to the material and the production of the component. Furthermore, these data, particularly for lead cracks, are characterized by exponential crack growth behaviour. Because of this common characteristic, the DSTO framework can use lead crack growth data to provide reasonable (i.e. not overly conservative) lower-bound estimates of typical crack growth lives of components, starting from small natural discontinuities and continuing up to crack sizes (thus encompassing short-to-long crack growth) that just meet the residual strength requirements. Scatter factors based on engineering judgement are then applied to these estimates to determine the maximum allowable service life (safe life limit).

The aim of the paper is to present the framework of assumptions and observations used in conjunction with a unique measure of the initiating discontinuity and a simple crack growth law to predict a lower bound fatigue life estimate.

1. Introduction

1.1. Fatigue life testing for metallic airframes

Accurate prediction of the fatigue lives of metallic airframes still presents challenges, particularly for high performance aircraft. There is always a demand for lighter structures with reduced manufacturing and operating costs. This leads to relatively highly stressed and highly efficient designs where fatigue issues can arise at features such as shallow radii at the junction of flanges, webs and stiffeners, as well as at holes and tight radii. As a consequence, there are usually many areas that need to be assessed for their fatigue lives, and many potential locations at which cracking may occur in-service.

It is well-known that fatigue is a complex phenomenon that is dependent on many parameters, including the material characteristics (mechanical properties, microstructure and inherent discontinuities, e.g. constituent particles), surface treatments and finishes, the component and structural geometries, dynamic load histories and the environment. Nevertheless, engineering fatigue design relies in-part on baseline coupon tests to assess the many locations identified as susceptible to cracking. The coupons may be loaded by constant amplitude (CA) or representative variable amplitude (VA) load histories, and they may try to represent some feature of a built-up structure. The results of these coupon tests are averaged to give an indication of the life of the structure in a production aircraft. However, there are significant limitations to this approach:

(1) Experience has shown that, in high performance aircraft, the structural components have many features with the potential to crack, and that each of these features is typical of a single type of (more-or-less) representative coupons. Hence, the average indicated life of a component is equivalent to only the shortest average life from tests on several types of coupons.
(2) Even when the most critical feature of a component has been identified and assessed by coupon testing, the coupons are rarely fully representative, notably with respect to the surface treatments and finishes required for production aircraft.

* Corresponding author. Tel.: +61 3 9626 7653; fax: +61 3 9626 7089.
E-mail address: Lorrie.Molent@defence.gov.au (L. Molent).

0142-1123/$ - see front matter Crown Copyright © 2010 Published by Elsevier Ltd. All rights reserved.
doi:10.1016/j.ijfatigue.2010.09.009

Nomenclature

a	crack size
a_0	initial discontinuity or pre-crack size
a_{cr}	critical crack size
a_{RS}	critical crack size at 1.2DLL
AA	aluminium alloy
AFHRS	airframe hours
BLKHD	bulkhead
β	finite width and crack shape geometry correction factor
CA	constant amplitude
CS	central spar
DLL	Design Limit Load
DSTO	Defence Science and Technology Organisation
DT	Damage Tolerance
EPS	equivalent pre-crack size
FASS	forward auxiliary spar station
FCG	fatigue crack growth
FLEI	fatigue life expended index
FSFT	full-scale fatigue test
IPP	inner pivoting pylon
λ	constant in specific exponential crack growth relations
LHS, RHS	left hand side and right hand side
N	time in specific exponential crack growth relations
NATO	North Atlantic Treaty Organisation
NDI	Non-Destructive Inspection
QF	quantitative fractography
R	stress ratio, S_{min}/S_{max}
RAAF	Royal Australian Air Force
RS	residual strength; also rear spar
RST	residual strength test
RTO	Research and Technology Organisation
S_{max}, S_{min}	maximum and minimum stresses
SF	scatter factor
SFH	simulated flight hours
SLL	safe life limit
t	time
USAF	United States Air Force
VA	variable amplitude

This is important because the commencement of fatigue cracking is primarily surface-influenced and therefore greatly dependent on small surface discontinuities inherent to component production, as well as any surface-connected discontinuities inherent to the material.

These limitations are addressed by other means. One way, which is mandatory for all modern aircraft, is to test actual components, part of the structure or even the full airframe, thereby including the effects of component geometry and production. Another way is to improve coupon testing by making the coupons optimally representative of the most fatigue-critical details, e.g. by applying surface treatments and finishes used in component production. This may seem obvious, but it is sometimes neglected or overlooked.

1.2. Fatigue lifing methods

1.2.1. Royal Australian Air Force (RAAF) lifing criteria

The RAAF methodology for lifing aircraft primary structures, e.g. [1], requires establishing the fatigue test life, under representative loading, of a full-scale structure or major component to a residual strength (RS) requirement of 1.2× Design Limit Load (DLL) without failure. Whether the test structure fails below 1.2DLL or survives, it is necessary to determine the equivalent fatigue life defined by the ability of a structural detail to achieve and survive 1.2DLL with cracking present. In other words, the test time to the critical crack length/depth (a_{RS}) at the RS $\geqslant$ 1.2DLL point is required.

For a crack that fails the structure below 1.2DLL the fatigue crack growth (FCG) life is assessed analytically and reduced to a time at which it would have reached the calculated a_{RS} value for a RS = 1.2DLL. For those cracks that survive the RS test load some assessment of the remaining amount of life may be needed. This depends on several factors:

(1) During a complex full-scale fatigue test, it is often necessary to ensure the survival of the test article by removing or modifying cracked locations when the cracks are smaller than the calculated a_{RS} values. These locations become the subject of fleet action prior to the overall life established by the fatigue test, but it may be possible to gain some additional life before the fleet action. This is checked by calculating the remaining FCG life to an a_{RS}, thereby establishing a virtual test life (virtual test point) for fleet action.

(2) Although the test may in general establish adequate fatigue lives, it is often not possible to apply representative load histories in all areas. When cracks form at locations in non-representatively loaded areas it may be necessary to calculate the definitive FCG life to a_{RS} and establish additional virtual test points. Such calculations require detailed knowledge of the FCG behaviour under representative and non-representative load histories.

(3) Finally, the load histories experienced by the fleet may turn out to be significantly different to the load histories assumed and applied during testing. As before, such differences may require further analysis of the cracks found during testing, in order to establish new equivalent test lives and virtual test points.

Each of these scenarios needs a framework of rules under which FCG predictions can be made with the aid of data from coupon, component and full-scale fatigue tests. However, before proceeding to this topic, which is the main theme of the present paper, methods of establishing the FCG lives are concisely discussed. This is because there is a major and fundamental difference between the method employed in the Damage Tolerance (DT) concept developed by the United States Air Force (USAF) [2] and the currently proposed and used DSTO method.

1.2.2. Methods of establishing FCG lives

Both the USAF DT and DSTO methods assume that defects (cracks, flaws and discontinuities) are already present in new structures, and that these defects must be treated as cracks that are immediately capable of growing by fatigue under service load histories. However, beyond these assumptions there are major and fundamental differences.

1.2.2.1. USAF DT method [2]. For critical locations the DT method specifies initial flaw/crack sizes and shapes based on pre-service Non-Destructive Inspection (NDI) capabilities and the assumption that cracks grow soon after the aircraft is introduced into service. The minimum assumed crack dimension is about 0.5 mm, see Table 1. Soon after these requirements were introduced, there