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FEM based fatigue crack growth predictions for spar of light aircraft under variable amplitude loading

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ABSTRACT

In the last decades, the assessment of the service durability of aerospace components and assemblies has become an important segment of design, mostly because of the growing needs for the light-weight structures which will be safe and reliable, and at the same time, not too expensive. It is especially true for the main structural elements such as wing spar, fuselage bulkheads and fittings, whose sudden failure could lead to the catastrophic consequences. In order to meet the strict safety requirements, as well as to check structural components before usage, a number of expensive and long experiments are carried out. Taking into account ever-present manufacturerś tendencies to shorten time-to-market periods, the use of finite element method (FEM) for the estimation of fatigue life has been proved as a good alternative to the experimental methods. The purpose of this article is to show that it is possible, by using finite element analysis (FEA), to obtain not only the good estimation of the fatigue life of the assembly such as the spar of the light aircraft, but also a good prediction of a number of load cycles which will propagate a crack on the spar to a certain length. On the basis of these results, it is possible to determine the proper inspections intervals which could prevent the catastrophic failure of the aircraft structure under variable amplitude loading.

1. Introduction

Standard specimens tested under the constant amplitude loading have given a number of data on how materials behave under different fatigue conditions [1,2]. However, during service period, structural components of aircrafts are subjected mostly to the variable amplitude loading. This implies that available fatigue data from tests under constant amplitudes cannot be applied without the appropriate modifications. Furthermore, the geometry of real component is very often significantly different from the geometry of the specimens, which could highly influence the accuracy of fatigue life predictions. In such a case, experimental verification of fatigue life of aircraft components under the variable amplitude loading must be carried out. But even the smallest change in loading, geometry or the materials of components, necessarily leads to new experiments, which makes the design process more expensive.

In a case when numerical model of the aircraft component (or assembly) is developed, all required changes are relatively easy to implement. Then, new estimations of the fatigue life can be obtained quickly and at low costs. However, the question that has to be answered is: how can we be sure that the numerical model is good, i.e. that obtained results are reliable and acceptably precise? It is obvious that an initial numerical model must be experimentally verified, and after that the values of fatigue life, obtained by calculations for variable amplitude loading, may be considered as good enough. Of course, the

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determination of absolutely accurate values is impossible because of the probabilistic nature of fatigue failures, i.e. identical components, subjected to the same loading spectrum during the experiment, very often experience large variations in fatigue life. However, numerical methods at least help us to define the criteria for either the acceptation or the rejection of the design and also allow Concept A to Concept B comparisons to be made without the need for high accuracy [3].

2. Material and methods

The main roles of the aircraft structures during its life are to transmit and carry loads and provide required lift force (mostly generated by airfoils moving through air). This is achieved by using thin-walled structures in which the interior surfaces are reinforced by longitudinal and transversal strengthening members. A predominant method used for joining these elements is riveting.

The wing represents one of the most important assembly zones on aircraft structures, meaning that special attention must be paid to the determination of the wing fatigue life and analysis of the fatigue failure [4]. The most important element of the wing is I-beam (called spar) which provides stability under compressive loads. It is extended along the length of the wing, in a direction perpendicular to the fuselage. Normally, the wing has 2–3 spars, but light aircrafts mostly have wing with one spar [5]. The spar carries almost all of the bending and shears loads, among of which the lift force is the most dominant. Generated lift force bends the whole wing upright; as a consequence, the upper elements of the spar are under compression, while the lower are under traction.

The subject of crack growth analysis in this case was I-beam of the light aircraft, shown in Fig. 1. A critical zone in which, under service loading, the crack is most likely to appear, is located in the wing root on the lower side and covers the spar caps and the wing skin. In this case, spar caps and spar web are connected with two rows of rivets of 3.2 mm in diameter.

Spar assembly used in fatigue testing, as well as washers and supporters used to fix spar and apply the load during the experiment, is shown in Fig. 2. The spar without adjoining skin was tested, so the reinforcing effect of the wing skin was not taken into consideration.

The spar parts are mainly made of hardened aluminum alloys. The main advantages of aluminum alloys are lightness, high specific strength and good corrosion resistance. Two mostly used alloys in aerospace applications are 7075-T6 and 2024-T3 [6,7]. The spar caps in Fig. 2 were made from 2024-T3 aluminum plates (thickness 1.6 mm), while the spar web was made from the same material, but different thickness (1 mm). For the purpose of fatigue testing, ten identical spars have been made. The length of the spars was 600 mm which equals the length of the constant chord section shown in Fig. 1. In order to measure strain in the zone where crack initiation was expected, several strain gauges were used. Strain gauges location is shown in Fig. 3.

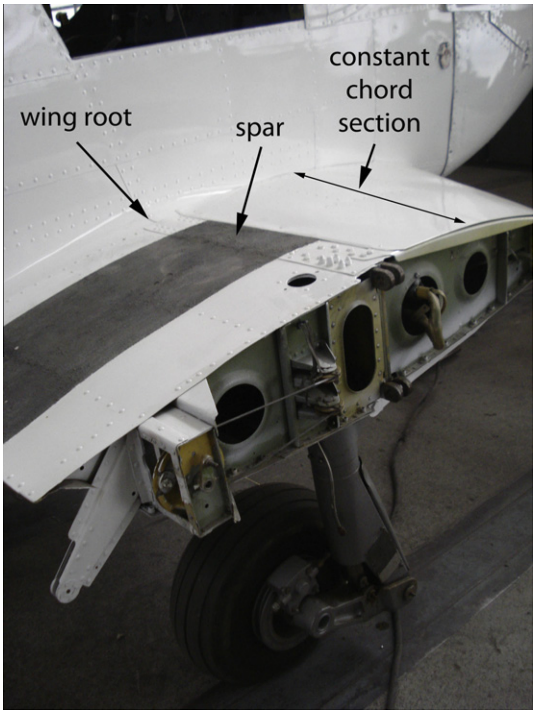

Fig. 1. Wing root assembly of the light aircraft.