

# Investigation of Fatigue Life of 2024-T3 Aluminum Spar Using Extended Finite Element Method (XFEM)

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## ABSTRACT

The assessment of the service durability of aerospace components and assemblies has become an important segment of design. In order to meet strict safety requirements, a number of complex and long experiments are carried out. The use of finite element method (FEM) and extended finite element method (XFEM) for the estimation of fatigue life and fatigue crack growth predictions has been proved as a good alternative to the expensive experimental methods.

In this paper, both experimental and numerical analyses of 2024-T3 aluminum spar of a light aircraft under variable amplitude loading are presented. FEM has been used for estimation of the spar life to crack initiation, whereas XFEM has been used for fatigue crack growth predictions and fatigue life estimation of damaged spar. The values of stress intensity factors were extracted from the XFEM solution in MorfeoCrack for Abaqus software. Numerical results (i.e. number of cycles to failure) have corresponded well with the experimental values obtained for the spar made of 2024-T3 aluminum alloy.

The investigations have shown that it's possible, using XFEM, to obtain not only the good estimation of 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 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.

**CITATION:** Rasuo, B., Grbovic, A., and Petrasinovic, D., "Investigation of Fatigue Life of 2024-T3 Aluminum Spar Using Extended Finite Element Method (XFEM)," *SAE Int. J. Aerosp.* 6(2):2013, doi:10.4271/2013-01-2143.

## 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 appropriate modifications. Furthermore, geometry of real component is very often significantly different from geometry of the specimen, 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 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 must 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 a good enough. Determination of absolutely accurate values is, of course, impossible because of the probabilistic nature of fatigue failures. However, numerical methods at least help designers to define the criteria for either the acceptance 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].

Today, aircraft engineers are making great efforts to analyze and predict fatigue crack growth in aircraft structures. It is widely accepted that the small crack propagation takes up the most of the fatigue life, especially in materials (like aluminum alloys) used for primary aircraft

components such as the wing spar [4, 5, 6]. Fatigue failure of wing spar may lead to catastrophic consequences, so the analysis of fatigue crack growth in spar under cyclic loading conditions is of the greatest importance.

In the past decades, several methods for crack growth predictions in various structures have been developed. The most important are: the boundary element method (BEM) [7, 8], mesh free methods [9, 10, 11] and finite element method (FEM) [12, 13, 14]. The extended finite element method (XFEM) is a new approach established in recent years [15, 16] and, in order to be widely accepted, it still has to be “verified” in practice. This means that crack growth predictions obtained for the complex 3D geometry using XFEM still cannot be accepted without experimental verifications. Calculations of wing spar fatigue life under variable loading presented in this paper should contribute to making more objective judgment of the usefulness of XFEM for crack growth predictions in aircraft structures.

## MATERIAL AND METHODS

The wing represents one of the most important assembly zones on aircraft structures and this is why special attention must be paid to the determination of the wing fatigue life and analysis of the fatigue failure [17]. One of the main parts of the wing is spar (or I-beam) which provides stability under compressive loads. The spar carries almost all of the bending and shears loads and during the flight upper areas of the spar are under compression, while lower areas are under traction.

The subject of crack growth analysis was I-beam of the light aircraft UTVA 75, shown in [Figure 1](#) and [Figure 2](#).



**Figure 1. Light aircraft UTVA 75**

A critical area 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 (diameter 3.2 mm).

The 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 [Figure 3](#). The spar without adjoining skin was tested, so the reinforcing effect of the wing skin was not taken into account.

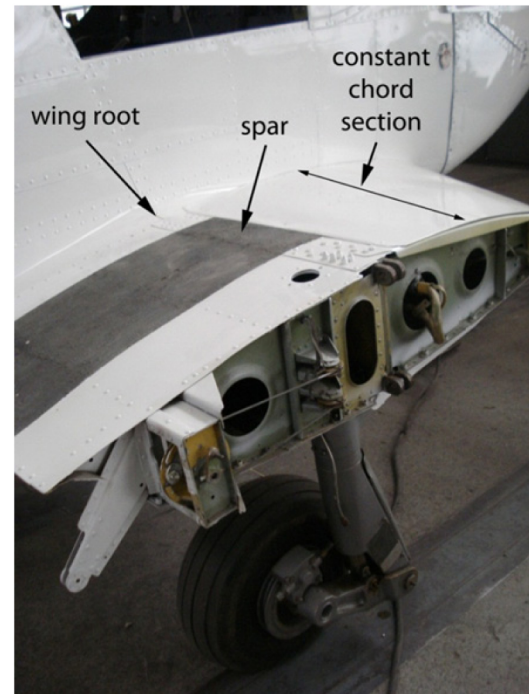


**Figure 2. Wing root assembly of the light aircraft UTVA 75**

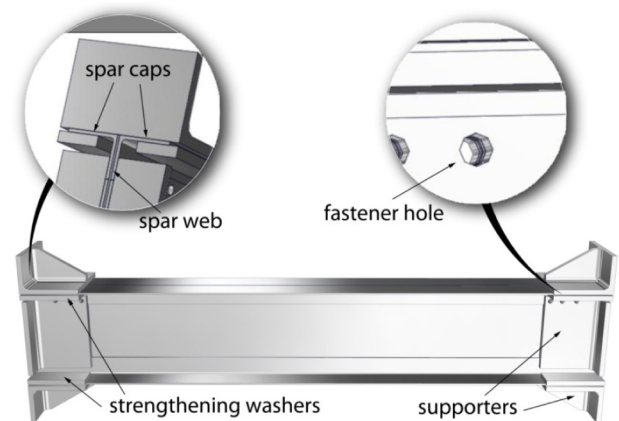


**Figure 3. The spar and supporting elements used in experiment**

The spar components are mainly made of hardened aluminum alloys. The spar caps in [Figure 2](#) were made from 2024-T3 aluminum plates (thickness 1.6mm), while the spar web was made from the same material, but different thickness (1mm). For the purpose of fatigue testing, ten identical spars have been made. The length of the spars was 600mm which equals the length of the constant chord section shown in [Figure 2](#). In order to measure strain in the zone where crack

initiation was expected, several strain gauges were used. Strain gauges locations are shown in [Figure 4](#).

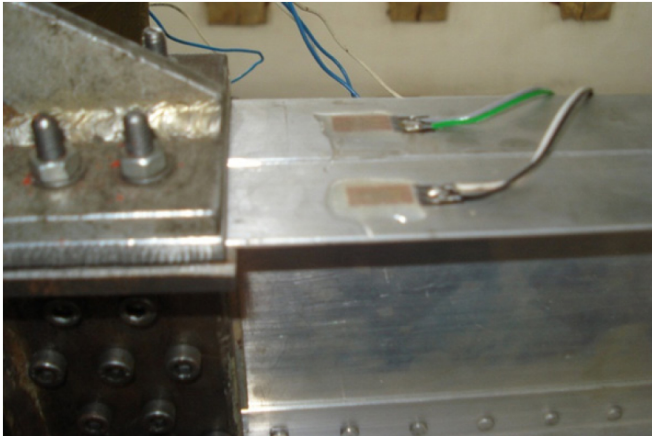


**Figure 4. Locations of strain gauges used in experiment**

One of the fundamental issues in the reliability analysis of fatigue behavior of real components under service loading is the evaluation of load or stress spectra. It is a well-known that the nature of the service load is stochastic and that data from original operating conditions are often not available [18, 19, 20, 21]. The aircraft service load history is a mixture of random loads and deterministic loads (such as loads in maneuvers). In the past, only deterministic loads were used in experiments, with the random loads represented by one or two amplitudes.

Several standardized load histories have been developed for the purpose of simplification and standardization of fatigue testing procedures. Different load histories have been defined for different types of aircrafts (TWIST and mini-TWIST for transport aircrafts testing, FALSTAF and mini-FALSTAF for fighter aircrafts, etc.) Available data may also be used for aircraft types with no defined standardized histories because the sequences of loads in standardized histories are completely defined in numerical format and the load scale is selectable. The main problem with the use of standardized spectra in the fatigue life estimations is the omission of numerous small cycles, because if these cycles were included, tests could take a very long time.

Hence, in order to get reliable values of the fatigue life, it is necessary to test the component or the whole assembly under the variable amplitude loading, by using either available service load history (from the aircraft of the same type) or using standardized spectrum.

## Fatigue Testing System Used in Experiment

The increased complexity of the load history that must be simulated requires development of expensive computer-controlled systems. Much simpler and a cheaper system (shown in [Figure 5](#) and [Figure 6](#)) has been developed for the purpose of fatigue testing of light aircraft spar. This system can simulate different and complex load histories by

changing the position of adjustable drawbar and configuration of adjustable eccentric disc [22, 23].

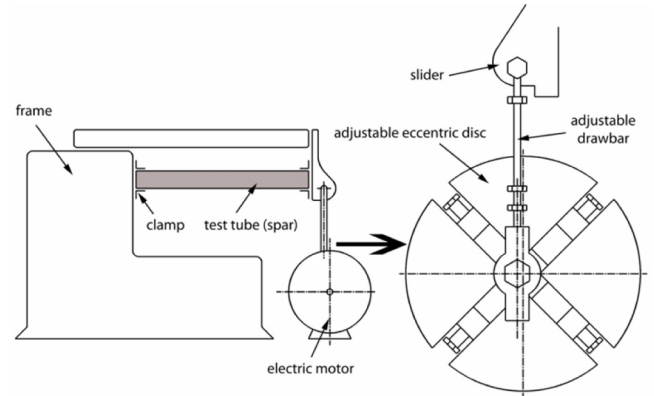


**Figure 5. Testing system used in experiment**

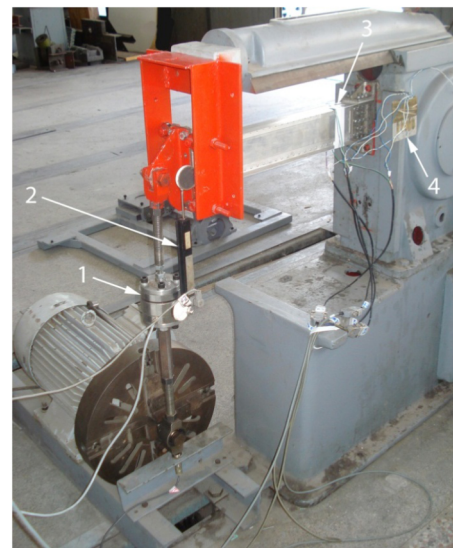


**Figure 6. Main components of fatigue testing system**

During experiment, the force on the drawbar is measured using the force transducer of type U3 50kN, manufactured by Hettinger Baldwin Messtechnik (Pos. 1 in [Fig. 5](#)). Considering the drawbar to be rigid under predicted external force, deflection of one end of the spar is measured relatively between the spar's end and center point of the drawbar, via potentiometer displacement transducer PM2S 150mm, manufactured by ELAP Spa (Pos. 2 in [Fig. 5](#)). Stresses are measured in the clamp area, at 4 points, using aluminum strain gauges of the type 1-LY13-10/120, produced by Hettinger Baldwin Messtechnik (Pos. 3 in [Fig. 5](#)). Each strain gauge is connected to one dummy strain gauge (Pos. 4 in [Fig. 5](#)) into one half of Wheatstone bridge circuit, and then with the acquisition system. Data acquisition and analysis are performed using SPIDER8 measurement acquisition system