Fatigue life of integral skin-stringer panels produced by laser beam welding (LBW) is analysed. This type of panel is used in airframe construction where fatigue and damage tolerance are of paramount importance, since aircrafts must be designed to tolerate relatively large fatigue cracks. The analysed integral skin-stringer panel is made of AL-AA 6156T6/2.8 mm, where stringers are laser beam welded to the airframe skin. By using the extended finite element method (XFEM), fatigue life of simple flat plate, as well of the skin-stringer panel, are numerically simulated in order to investigate the effect of stringers.

INTRODUCTION

Fatigue crack initiation and growth as well as fracture resistance and corrosion issues associated with riveted structures are well understood and it seems difficult to get significant improvements in riveting technology regarding the extension of the fuselage fatigue life. However, the development of welding technology and the integral skin-stringer structures have reduced riveted joint applications, /1/. Integral skin-stringer structures (Fig. 1), which make skin and stringers as a continuum, are more suitable for improvements even though they might be poor at damage tolerance performance. Compared to conventional riveted structures, integral skin-stringer structures have many advantages, such as lower weight and lower cost to manufacture. It is worthy of note that fewer components mean they are easier to inspect and, at the same time, integral structures have less holes with high stresses suitable for crack initiation. Aluminium alloys are major materials for light-weight constructions due to their good mechanical properties and low density, and their recent developments have led to the use of advanced welding technologies to reduce weight and fabrication costs, /2-5/. Laser beam welding (LBW) has been successfully applied for manufacturing skin-stringer curved panels for various civilian aircraft in Europe. Friction stir welding (FSW) is also considered as a prospective welding process for butt-joint applica-

Fig. 1. Integral skin-stringer structure of the fuselage.
hoop stresses (induced by cabin pressurization) and circum-
ferential cracks under stresses from vertical bending of the 
fuselage, /8/. A critical element of damage tolerant design 
in pressurized fuselage is the ability to predict the growth 
rate of fatigue cracks under applied loading.

The crack growth stage is studied by using the stress 
intensity factor (SIF). SIF is a fundamental quantity that 
governs the stress field near the crack tip. It depends on the 
geometrical configuration, crack size, and loading condi-
tions of the body. There are many methods used in numerical 
fracture mechanics for SIF calculation. FEM has been 
used for decades for calculating SIFs, but it has some 
restrictions in crack propagation simulations, mainly 
because the finite element mesh needs to be updated after 
each propagation step in order to track the crack path. 
Extended finite element method (XFEM) suppresses the 
need to mesh and remesh the crack surfaces and is used for 
modelling different discontinuities in 1D, 2D and 3D 
domains. XFEM allows for discontinuities to be repre-
sented independently of the FE mesh by exploiting the 
Partition of Unity Finite Element Method (PUFEM), /9/. In 
this method, additional functions (commonly referred to as 
enrichment functions) can be added to the displacement 
approximation as long as the partition of unity is satisfied. 
XFEM uses these enrichment functions as a tool to repre-
sent a non-smooth behaviour of field variables. There are 
many enrichment functions for a variety of problems in 
areas including cracks, dislocations, grain boundaries and 
phase interfaces. Recently, XFEM and its coupling with level 
set method are intensively studied. The level set method 
allows for treatment of internal boundaries and interfaces 
without any explicit treatment of interface geometry.

NUMERICAL SIMULATIONS USING XFEM

The main idea of numerical modelling was, firstly, to 
test XFEM by making FE model of base metal plate with 
initial crack, as shown in Fig. 2a, simulating the real loads 
from experiment, and consequently comparing the number 
of cycles obtained numerically to the number of cycles 
obtained experimentally. Base metal plate is chosen as it 
had simple geometry, and because the calculated values of 
SIF could be verified using other methods, or can be even 
found in the literature. The second step was FE modelling 
of 4-stringer panel (Fig. 2b) and determination of the 
number of load cycles that would grow crack to critical 
length that will then be compared to number of cycles 
obtained in experiment with real 4-stringer structure. In 
both simulations the aluminium alloy AA6156 T6 is used 
(Young’s modulus \( E = 71000 \) MPa, Poisson’s ratio \( \nu = 0.33 \)), and the FE models of base plate and 4-stringer plate 
are shown in Fig. 2. The loads used in simulations equal to 
the average values of maximum tensile forces over time 
tested in experiments, /6/. For base metal plate, the 
average maximum force is \( F_{\text{max}} = 112.954 \) kN, while the 
load ratio \( R = 0.146 \) is determined on the basis of average 
minimum tensile force measured. Coefficients for Paris 
equations are adopted on the basis of the values obtained in 
tests with base metal plates: \( m = 3.174 \) and \( C = 1.77195^{12} \) 
MPa·mm\(^{1/2}\).

Figure 2. FE model of base metal plate with initial crack, a) simple 
panel, base metal, b) integral 4-stringer panel.

Initial crack in the first simulation propagated to length 
\( 2a = 275 \) mm, and Fig. 3 shows its shape after the last 
growth step. As it can be seen in Fig. 4, the number of 
cycles predicted by Paris equation incorporated into 
Morfeo/Crack for Abaqus software is comparable to the 
number of cycles obtained in one of the experiments with 
base metal plate (different values of number of cycles are 
obtained in series of experiments; however, the deviation is 
not above 15%).

Figure 3. Crack in base metal plate after 260 steps of propagation 
(\( 2a = 275 \) mm).

Figure 4 shows that in the XFEM simulation, the number of 
cycles to critical crack length is less than that obtained in 
experiment (169 076 cycles versus 189 514 cycles, which is 
a difference of about 10%); however, under crack length 
\( 2a = 60 \) mm (almost linear growth) the numbers of cycles 
differ insignificantly. This confirms the previously drawn 
conclusion, /10/, that in case of simple geometry, XFEM is 
a fairly reliable method for crack growth rate determination, 
as it provides more conservative values compared to the 
experimental ones.
After successful numerical simulation of crack growth on base metal plate, a more complex geometry of the 4-stringer plate is analysed (Fig. 2a). The central crack of length \(a_0 = 14\) mm is initiated and the load identical to that used for the base metal plate is applied. The crack propagated for a total of 173 steps (in each step the crack length increased by 2 mm). During the 160-th step, complete failure of the left stringer occurred (Fig. 5), after which the crack continued to spread along the right stringer and through the base metal plate. Simulation of crack growth stopped after 173 steps because the number of load cycles necessary to propagate the crack by one millimetre dropped under 100, which is a sign that the crack started to propagate rapidly and that the 4-stringer plate is under complete failure.

![Figure 4: Numbers of cycles obtained in experiment and XFEM simulation (base metal T6).](image1)

![Figure 5: Crack after 160 steps of propagation.](image2)

Finally, the crack length vs. number of cycles is shown in Fig. 6, as obtained in simulation with 4-stringer plate and compared to experimental results, in the same way as in the case of the base metal panel, Fig. 4.

![Figure 6: Number of cycles for base metal plate obtained in the experiment and 4-stringer plate obtained in simulation by XFEM.](image3)
DISCUSSION AND CONCLUSIONS

The estimation of fatigue life of integral aircraft structures, being the subject of this paper, is also very important although the possibility of crack occurrence in the integral structure is lower than in the case of a differential structure. XFEM can play a significant role in this subject and may reduce to a minimum the number of experimental verifications and so fulfil one of the requirements mentioned at the beginning of this paper: cost-efficient design. The results of numerical simulations based on XFEM presented in this paper are quite well correlated with experimental values, which is particularly true for the simulation of crack propagation in a 4-stringer plate.

There are, of course, some differences due to the fact that the initial length of the crack in experiment is \(2a_0 = 75\) mm whereas in simulation \(2a_0 = 14\) mm, because the intention was to compare the 4-stringer plate with the base metal plate whose initial crack length was \(2a_0 = 14\) mm; as a result, the graph presented in Fig. 6 is created. Figure 6 shows that the 4-stringer plate with given stringer dimensions and for given tension force has a fatigue life approximately 30% longer than the base metal plate (254 274 cycles vs. 194 453 cycles), estimated on the basis of the Paris law (unfortunately, the information on the number of cycles obtained in experiment with the 4-stringer plate is not available). There are other equations that can be used to estimate fatigue life under given conditions (NASGRO equation, for instance) and they might give another number of cycles, but not so different from the one presented here. If aircraft designers consider the improvement by 30% as not satisfactory, the redesign of a 4-stringer panel and consequent tests on new panels would take much time and money. However, using XFEM estimation of the total fatigue life of a new structure is relatively easier to obtain and inexpensive, not only for loads used in experiments but for the whole spectrum of loads that might appear during the intended life of the structure. So, this is the major competitive advantage of XFEM.

REFERENCES