LARGE DAMAGE RESIDUAL STRENGTH ANALYSIS: SIMULATIONS AND TESTS

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Abstract

On some occasions the aeronautical composites structures suffer damages that put in danger the return home of the aircraft. The aim of this project is analyzing the behaviour of a composite structure with big damages, studying both initiation and progression of the mentioned damages. Detailed simulations have been made with the commercial finite elements code Abaqus and five curved stiffened panels with different damages have been tested in tension. After analyzing and comparing the results, it has been verified that the numerical predictions and the experimental results correlate adequately. Finally, a fuselage architecture has been validated.

1. Introduction

Contra-Rotating Open Rotor (CROR) is a promising technology for reducing fuel consumption of future aircrafts. One of the most critical challenges of such innovative engines is the safety/certification requirements in the case of blade release. Such high energy debris could be released and impact the aircraft, producing large damage and challenging the structure integrity, so it is needed to assure the safe continuation of flight and landing.

The certification guidelines about Damage Tolerance (Discrete Source Damage) evaluation [1] establish that the remaining structure after the incident has to support, with an acceptable level of confidence, 70% of the limit flight maneuver loads and, separately, 40% of the limit gust velocity (vertical and lateral) at the specified speeds. Besides, the stiffness has to be sufficient to avoid the flutter and to secure minimum handling qualities.

In order to secure the residual strength of the airframe structure made of innovative composite materials, after such high energy impact, it is necessary to perform large scale blade impact and residual strength test. To be able to demonstrate the feasibility of structure concepts at fuselage level, it is necessary to develop predictive simulations correlated at lower test level. To be able to achieve such accurate predictions, it is mandatory to be able not only to simulate the initiation of the damage progression but also the final progression failure of the structure.

There are several studies about residual strength prediction. Some of them [2] [3] [4] show models of stiffened panels with discrete source damages.

The main objective of this paper is to present the development of detailed nonlinear simulation models and methodology correlated by test, aiming to predict damage propagation initiation and final progressive failure of airframe composite structure, at full scale, after blade impact damage. First, discrete source damage behavior was analyzed at panel level, afterwards, the studies were extended at fuselage level.

2. Description of the analysis

After studying the most adequate methodology to develop this type of simulations, residual strength simulations and tests were carried out with 2m x 2m composite curved stiffened panels with 8 composite stringers and 4 metal frames. Theoretical damages are artificially generated, and the radius tip was defined as a half of the damage width.

Five panels were simulated with nonlinear finite elements models and tested in tension until ultimate rupture. Reference panel (called panel 1) had a damage that cuts three stringers and a width of 10mm; so the tip radius was R_1=5mm. Panel 2 and panel 3 were similar to panel 1 but they present a damage of 100mm and 380mm of width respectively, so their radius at the tip damage are R_2=50mm and R_3=190mm. The damage in panel 4 had the same width as in panel 1 but in this case the length of the damage was longer, having 4 damaged stringers instead of 3 (Figure 1). Finally panel 5 presented the same damage as panel 1 but having two aluminium stringer profile riveted close to the damage edges as design precaution to restrain the damage progression (Figure 2).

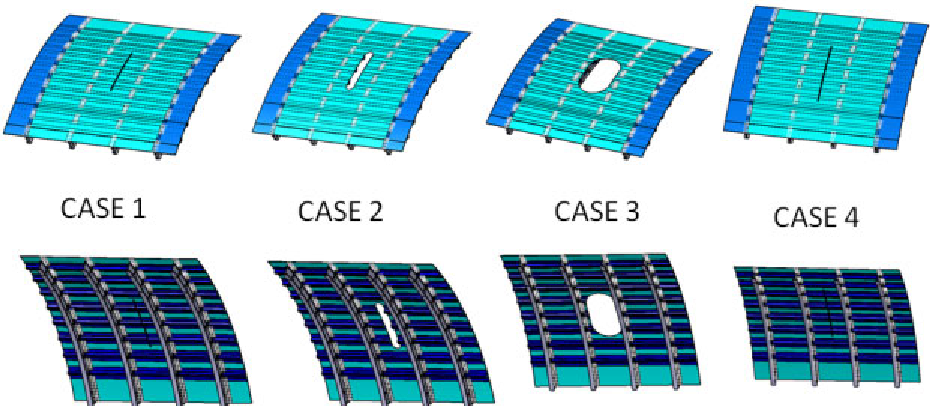

Figure 1. Cases 1, 2, 3 and 4. The difference between these four panels is related only with the damage geometry.

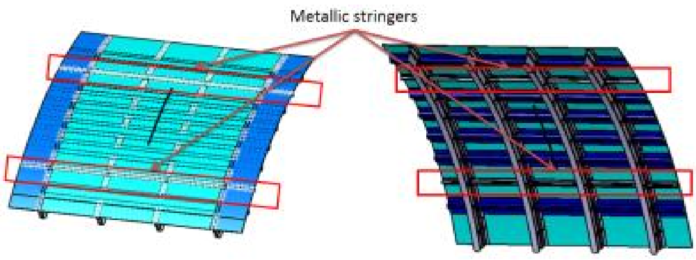

Figure 2. Panel 5 is similar to panel 1 (same damage geometry) but using metallic stringer for stringers 2 and 7 as design precaution.

3. Simulation and test results

3.1 Test definition

The five panels were tested applying tensile load. Loading and boundary conditions are represented in Figure 3.

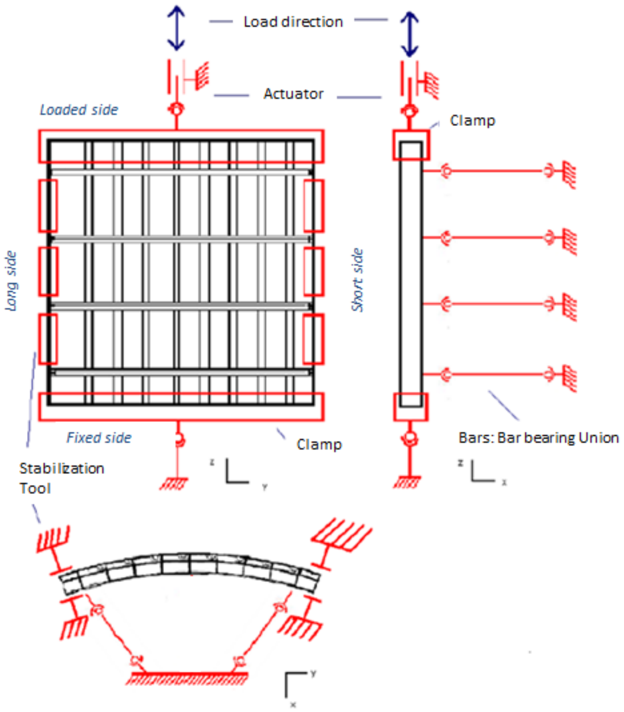

Figure 3. Tensile test diagram with load and boundary conditions

3.2 Modelization

Detailed non-linear models of the five panels were carried out with Abaqus finite elements code. The crack propagation path was dependent of the mesh and a reduced element size is needed near the tip damage. Hashin´s Initiation Criteria [5] were used with Damage Evolution to study the crack propagation in the composite, so the non-linear models included fiber and matrix material degradation.

The panels were modelized with Abaqus Explicit with mass scaling (a small value that does not have influence in the results) in order to reduce computation time.

3.3 Results comparison and correlation

The simulation results revealed a high stress gradient near the damage tip, as it is shown in Figure 4. The crack growth initiation depends on the damage tip radius, because it is the main factor driving the stress concentration.

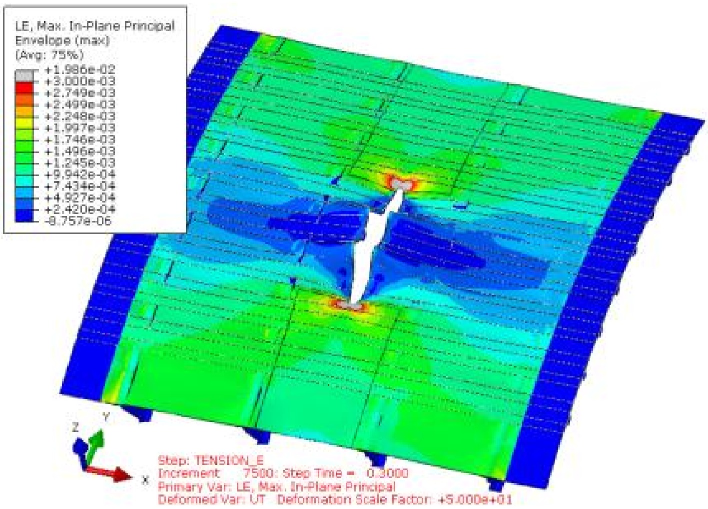

Figure 4. High stress gradient near the damage edge

The correlation between the crack growth predictions and the damage progression results obtained from experiments was very accurate. Figure 5 and Figure 6 show that the gauges measurements are similar to the predictions. Near the damage there were more than 15000 micro-strains and, in all the cases, the gauges in the far field reached more than 3000 micro-strains, so these values were higher than damage tolerance allowables.

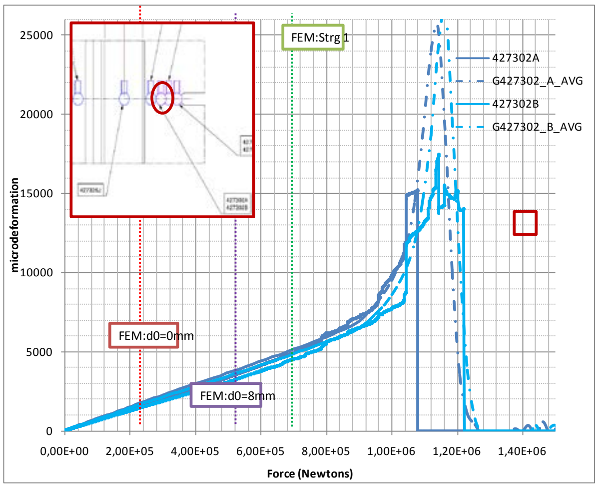

Figure 5. Representative strain near the tip damage

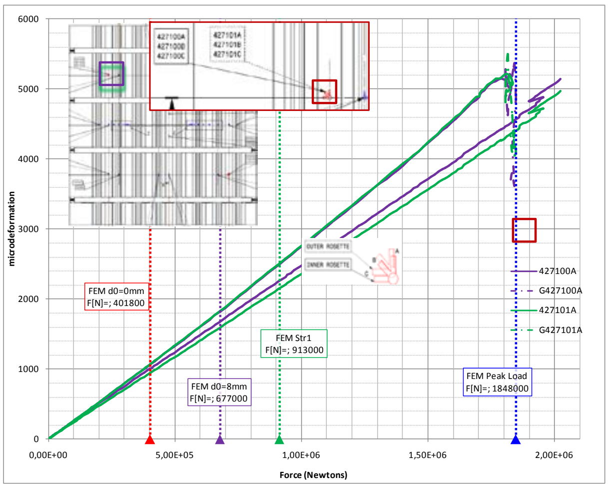

Figure 6. Representative strain far the damage

Figure 7 shows both the test and FEM simulation damage initiation load and final failure load for each panel. The panel net section failure without stress concentration is also marked in the graphic.

The results show that panels with lower radius ($R_1 < R_2 < R_3$) suffered the damage progression beginning earlier, because there was higher stress concentration at the damage tip. The final residual strength of these panels was similar and it was close to net residual section failure. Results comparison between panel 1 and 4 showed predicted decrease of residual strength with the damage length. Finally, results comparison between panel 1 and panel 5 showed that damage progression initiation load and final failure load were equivalent, so design precautions were not very effective. The panels could sustain important amount of load after initiation of failure progression until final failure.

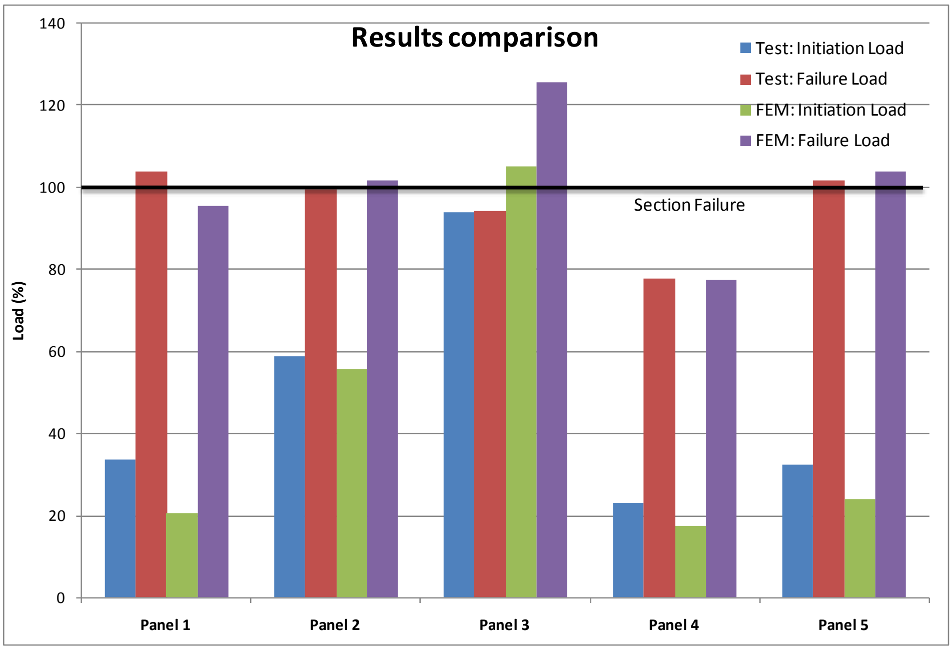

Figure 7. Load comparison between simulation and test for each panel

These results reveal that there is a stress alleviation mechanism at the edge of the damage. That means an important opportunity, because in very sharped damages, damage initiation and final failure were very far apart. Final failure loads resulting independent of this parameter and close to net section one.

It has been verified that composites do not have necessarily an explosive damage progression, and could support even four times more load once the progression has begun. Provided that when the damage grows the panel continues supporting load, it is possible that the damage stops after a small progression.

4. Application in a possible fuselage architecture

4.1. Description of simulation models at fuselage section level and previous studies

After testing and simulating at panel level, preliminary analysis of residual strength was performed at fuselage level in order to have more representative boundary conditions and loading introduction. Detailed finite elements models of the damaged structure fuselage section were generated. This detailed section was integrated in an aircraft structure global simplified Nastran Finite Element model to capture properly boundary conditions and load cases introduction. Get home loads and blade release imbalance loads cases were considered according to the certification requirements [1]. RBE3 elements were used to joint fuselage detailed section model to global FEM model, as it is shown in Figure 8.

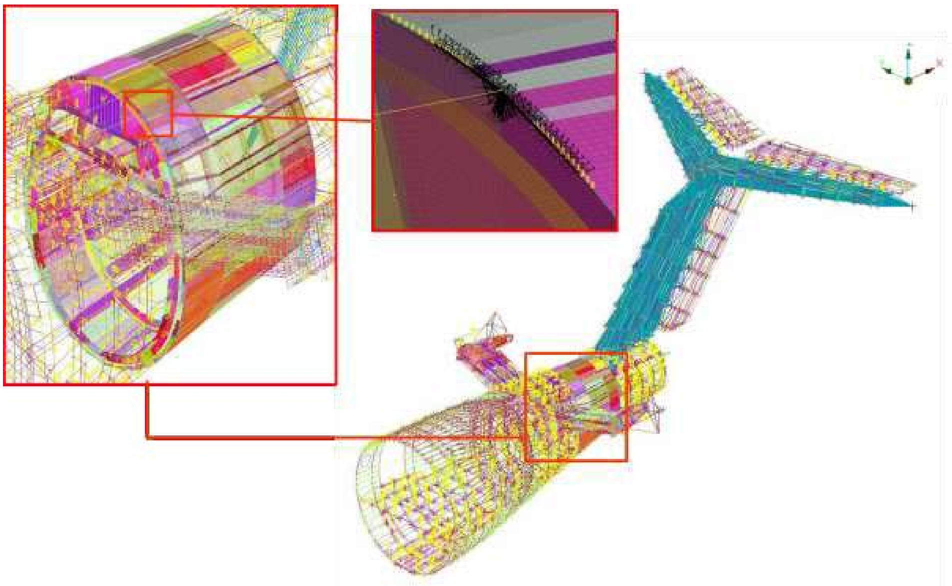

Figure 8. Detailed Finite Element Model at fuselage level

Damage that was consider in this model was determined after evaluation of possible blade trajectories. Enveloping clean cuts were analyzed to obtain the most critical damage positions. After that, preliminary detailed linear Nastran simulations were carried out. Six different forms of large damage were simulated. As on panel level, local mesh refinement was required around the damage to capture the stress concentrations induced. The ply failure criteria used was Yamada-Sun for fibre and Puck for matrix failure. These damage types were analysed for every required load cases and strength Reserve Factors were calculated, to determine the most critical damage and critical load case.

The results revealed a very earlier damage progression initiation in the sharper damages, like the analysis at panel level had shown, but provided no information about nonlinear progression and final failure.

This identified most critical damage consists of a clean circumferential cut in the upper part of the fuselage. This damaged structure with its critical load case was analysed by a non-linear progressive failure study in Abaqus Standard. In order to perform this type of stress analysis, the detailed Nastran sub-model containing the damaged structure was translated to Abaqus. The Abaqus sub-model was loaded in displacement mode in order to achieve solution convergence analysing the post first ply failure state.

3.2. Simulation analysis and results at fuselage section level

The Abaqus progressive failure analysis was performed on a detailed nonlinear sub-model of the critical damage fuselage section using applied displacements from the critical load case of the Nastran linear model. This non-linear model included fibre and matrix material degradation to model crack propagation in the composite in order to understand the initiation and final failure of the structure.

The damage propagation results are showed on Figure 9. It presents the damage propagation initiation in both sides of the damage and the subsequent progression of damage until the maximum load that the structure has to support.

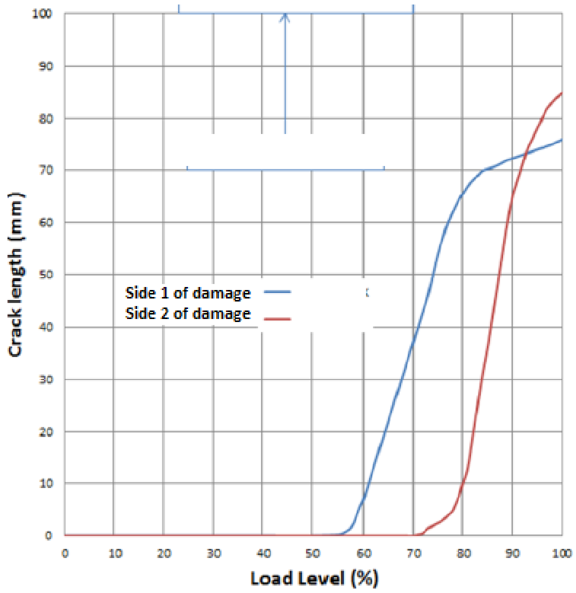

Figure 9. Damage crack growth for the critical load case

It is observed that damage progression initiates at side 1 and damage progression velocity increases with load until it is nearer to the reinforcement area; then, damage progression is slowed and does not reach the catastrophic failure. On the other side of the damage, the damage propagation initiated later and progressed until it also reaches the proximity of the adjacent reinforcement area, when the progression of damage with load is also significantly lowered down. Final rupture would be performed at a higher level than the required design load, validating the residual strength of the structural part.

To validate a determined architecture, the damage growth initiation should be studied. If there is no initiation for the most critical case, the configuration is validated but it is possible to optimize the design, reducing weight. On the other hand, if the damage progression starts and progresses fast to final failure, it is necessary to redesign the architecture and increase reinforcement and weight, because it is not capable to support the load requirements. Another possibility is that the damage progression starts but the growth is significantly decelerated and stopped before reaching the catastrophic failure. These scenarios are shown in the diagram of Figure 10.

In the presented study it was observed that, for the critical case, the defined structure behaviour is like on this third scenario. As we could consider that initiation is not final failure, the fuselage architecture defined could be considered as validated.

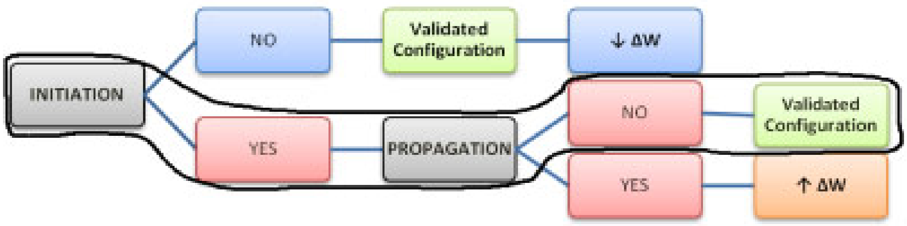

Figure 10. Architecture validation

5. Conclusions

Panel non-linear models were developed to understand the damage progression behaviour and compared with the tensile tests that were carried out. The results present a good correlation. Smaller edge radius produced earlier initiation of damage progression because there is a higher stress concentration. A significant difference is observed between damage progression initiation and final failure load, so damage progression is not necessarily explosive.

The damage propagation in composite materials may not be catastrophic. If damage initiation is allowed, final failure is similar to net section failure, so slow growth or damage arrest strategy could be a possible opportunity.

Linear detailed Nastran models were carried out at fuselage level to obtain the critical damage and load case, to be studied with damage progression. After that, non-linear progressive failure analysis with Abaqus was made to simulate the damage propagation and final failure of the structure.

Equivalent conclusion than at panel test and simulations is obtained. Damage progression initiated early, but its progression stopped with the required loads (GHL) before the failure was catastrophic. The damage growth was not explosive so the architecture was validated. This conclusion provides a preliminary verification at fuselage level that slow growth or damage arrest strategy could be applied to reduce reinforcing penalty weight.

Further maturation of these full scale models, including correlation with additional lower level test, with other loading cases such as compression, shear and pressure, should allow performing fully predictive virtual test to support and minimise cost of final full scale residual strength physical test.

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