Manufacturing process simulation and structural evaluation of grid stiffened composite structures

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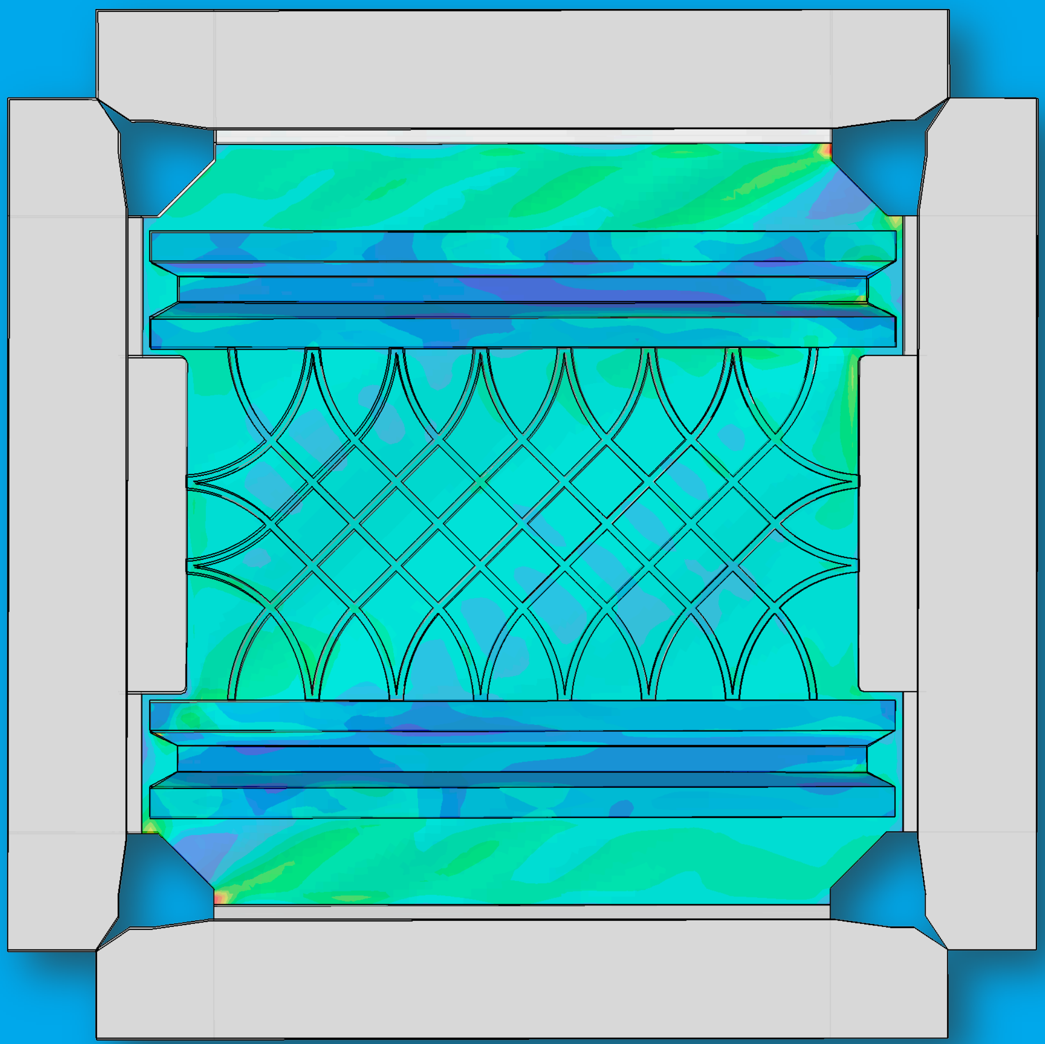

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EXECUTIVE SUMMARY

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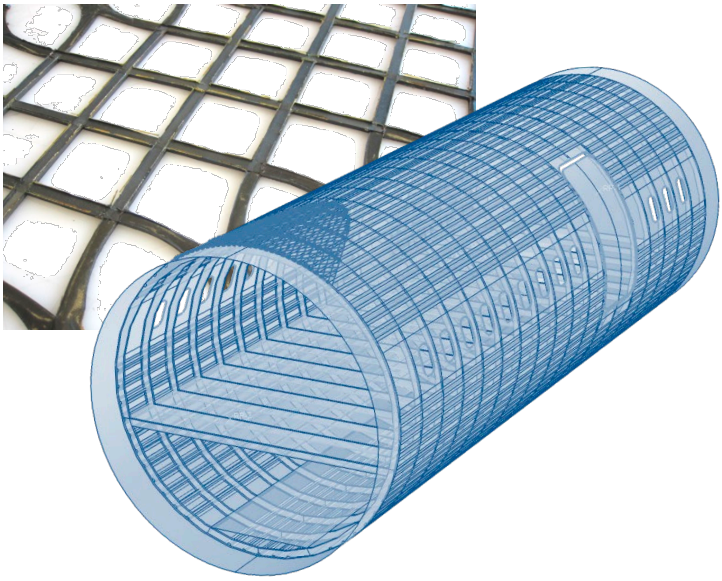

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Computational Mechanics and Simulation Technology
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Problem area

In recent civil aircraft the share of composite material has increased up to 50% of the total structural weight. In the EU MAAXIMUS project the forward fuselage of a single aisle airliner has been investigated and it was observed that the forward crown section was too heavy. Therefore innovative solutions such as grid stiffening have been investigated to see whether a benefit could be achieved in this area of the structure. From this research in the field of innovative composite structures new opportunities may arise for the Dutch aerospace industry.

Description of work

The fuselage section as shown in the figure above was analysed for a large number of loadcases. Based on this a single panel design using grid stiffening was created. The grid structure is a

This report is based on a presentation held at the Royal Aircraft Structures Conference, Belfast on the 7th of October, 2014.

local reinforcement of the skin that increases the buckling resistance without a high weight penalty. Follow up nonlinear analyses were performed to investigate the post-buckling performance of the grid stiffened structure. Damage tolerance was investigated using progressive damage methods to assess the sensitivity toward initial damage in a qualitative manner. To investigate the manufacturing effects of including a grid section on top of the skin was performed using an in-house developed curing model. The focus for this work was on a shear panel that was manufactured and tested.

Results and conclusions

The overall aim of this study was to evaluate buckling and strength performance of a single panel using numerical methods, taking into account composite curing aspects, manufacturing and testing of this optimized design.
The final weight gain is a theoretical 8.4 kilograms for the forward fuselage structure. This amounts to approximately 1% of the total structural weight. Analyses showed a stable post-buckling behaviour for the panel and a damage tolerant structure. The curing analysis however showed large residual stresses and distortions therefore preventive measures were taken in the panel manufacturing process itself.

Applicability

The grid stiffening design allows for a wide variety of applications, such as wing and stabilizer components. In the current research the entire design space for this structure has not been investigated. For instance a local variation of grid height can further improve the efficiency of the design. Also the damage tolerance analysis shows that the grid structure offers a redundancy due to the multiple possible load paths that can be used throughout the aircraft structure.

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Summary

This paper presents a study of carbon composite structures with advanced architectures. These architectures are specifically tailored for light weight aircraft components by application of integrated local skin reinforcement with the use of rib and grid stiffening. The overall aim of this study is to evaluate buckling and strength performance using numerical methods, taking into account composite curing aspects, manufacturing and testing of this optimized design.

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Content

Abbreviations

α	Cure state in analysis
B31	Beam element linear 2 node
DCB	Double Cantilever Beam test
ENF	End Notch Fracture test
GIc	Fracture toughness interface (peel mode)
LL	Limit Load
LVDT	Displacement measurement device during testing
mm	millimetre
RF	Reserve Factor
S4R	Shell element linear 4 node
T	Temperature in cure analysis
Tg	Glass transition temperature
UL	Ultimate Load

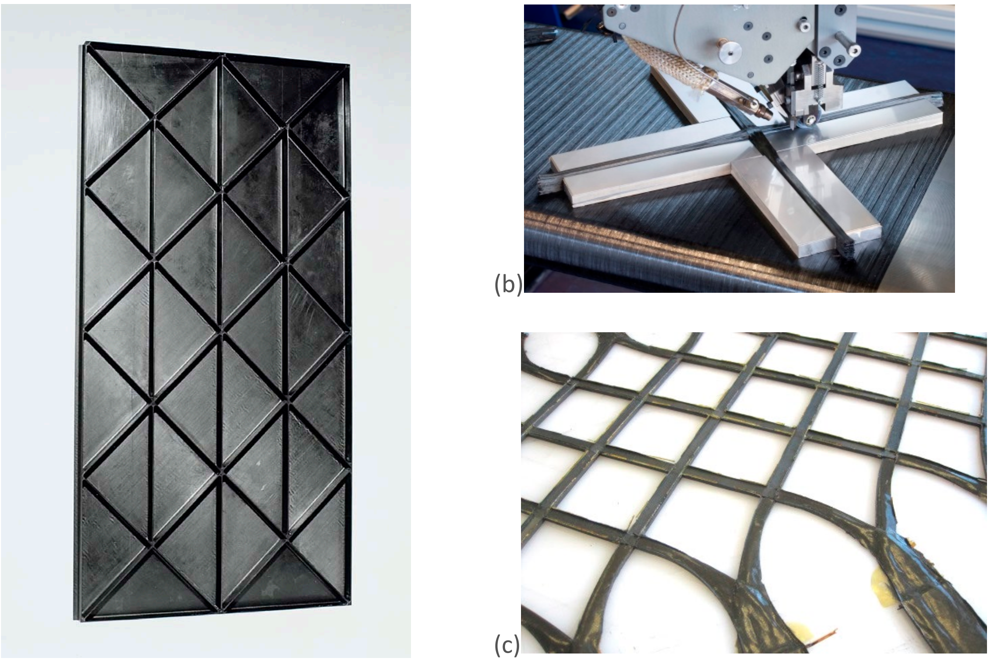
1 Abstract

This paper presents a study of carbon composite structures with advanced architectures. These architectures are specifically tailored for light weight aircraft components by application of integrated local skin reinforcement with the use of rib and grid stiffening. The overall aim of this study is to evaluate buckling and strength performance using numerical methods, taking into account composite curing aspects, manufacturing and testing of this optimized design.

On a coupon scale the effects of variations in grid height, grid/skin interface design and skin thickness have been analysed and manufacturing trials have been performed. The knowledge from these coupon level simulations and manufacturing trials have been used to design and manufacture an aircraft front-fuselage panel section with a buckling-optimized curved grid stiffening reinforcement. Tow cutting is used at each of the grid intersections to prevent tow overlapping and maintain quality of the laminate in the grid. In the investigated application area, grid stiffening designs in the front-fuselage section have shown weight gains compared to conventionally stiffened panel designs.

2 Introduction

Grid stiffened structures have been studied extensively in the past for space and aeronautic applications. At NLR this technology has received attention since the mid-19-eighties and several innovative concepts have been designed, manufactured and tested. One example is a flat composite panel using stiffening ribs in a triangular pattern. This panel was optimized for global and local buckling performance with minimum structural weight. More recent work was mainly focused on understanding the interface behaviour and cross-section design for grid stiffened structures. In Figure 1 an overview is given of previous and present work at NLR on grid stiffened structures.



(a) (b) (c)

Figure 1: Overview of some selected grid stiffened structures designed and manufactured at the NLR. (a) Grid stiffened panel using triangular shapes. (b) Production of a grid cross section using fibre placement technology. (c) Novel grid structure part of this investigation

Van den Brink et al. [1] presented a study where the grid design was optimized using several design parameters for grid angle, grid offset and grid height. The application of the grid stiffening was aimed at the relatively low loaded forward fuselage of a civil aircraft, see Figure 2.

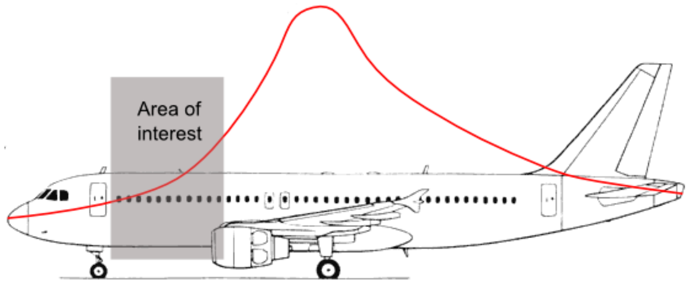


Figure 2: Civil aircraft with an indication of the bending loads in the fuselage and the envisaged application area of grid stiffened structures in the forward fuselage. In the forward fuselage the loads are relatively low and here innovative lighter stiffening solutions can thus be applied

Grid stiffening structures have been investigated in the past with variations on the application type and load distributions in the structure. In most studies the grid structure itself becomes a load carrying structure. In this study the grid stiffening is employed as an additional local laminate thickening to optimize the in-plane and bending stiffness of the skin achieving minimal weight. For metal structures the grid stiffening using iso-grids has been proven technology as shown in work by Huybrechts [2]. For composite structures the contribution by Vasiliev [3] and Wegner [4] in this field of grid structures is interesting from an application standpoint. In most literature the main downsides of composite grid stiffening structures are found to be related to the complex manufacturing process. Issues with the interface of the underlying laminate were observed by Baker [5] and Buragohain [6]. More recent studies towards replacing the aircraft fuselage frame and stringer topology with grid structures have been performed in the EU Alasca project [7]. From a numerical method perspective the research in the field of grid stiffening has been mainly focused on optimization of stiffness on a global level without including manufacturing aspects or residual strength. In work by Kidane [8] and Kanou [9] the numerical side of the novel structure has been investigated on a global level.

Previous experience from NLR with grid structures using carbon fibres and thermoset resin also revealed issues during manufacturing and final product quality and strength [1]. The main issue is related to the interface between the grid and skin laminate due to their respective orthotropic thermal expansion properties. During curing the expansion differences have to be carried by this grid/skin interface and in some instances this can lead to complete failure of the interface. Another effect of the stiffness difference is a distortion of the global structure that can be detrimental for the buckling performance in particular because of larger imperfections. Also this part distortion will have an effect on the assembly tolerances of the part. These manufacturing related issues are studied in this research using progressive damage analysis, thermal analysis and a newly developed curing analysis.

The present paper describes a study, executed within the MAAXIMUS project [2], where optimized grid structures are designed for aircraft fuselage and detailed analyses and comparisons with test results are performed. The presented work focuses on the shear behaviour of the grid stiffened panel which occurs in the fuselage front fuselage in case of nose gear yaw. In this study the focus lies on numerical method developments enabling the prediction of interface damage and curing behaviour of the grid structure in detail. Specific challenges for the manufacturing of proper grid-skin interfaces are investigated.

3 Design and detailed numerical simulation

In the study the grid panel is formulated as 'use-case' including manufacturing and testing results. The loadcases from the fuselage level were translated to panel level, where a compression/tension panel was defined and a shear panel. In this work the grid shear panel model for final testing as shown in Figure 3 is discussed which is the result of a parameter variation and optimization process [1] and the main dimensions are discussed in this section. The grid section consists of fibre placed tow section with a width of 6.35mm (1/4 inch) and the grid height is 2 mm. The spacing between the grid section is 50 mm using alternative cutting at the intersection in order to cope with the minimal fibre placement tow length. The size of the panel and the lay-ups are shown in Table 1. The material properties of the Hexcel composite material used for the panel models are derived from UD coupon test data. This material data cannot be disclosed for publication because of confidentiality agreements.

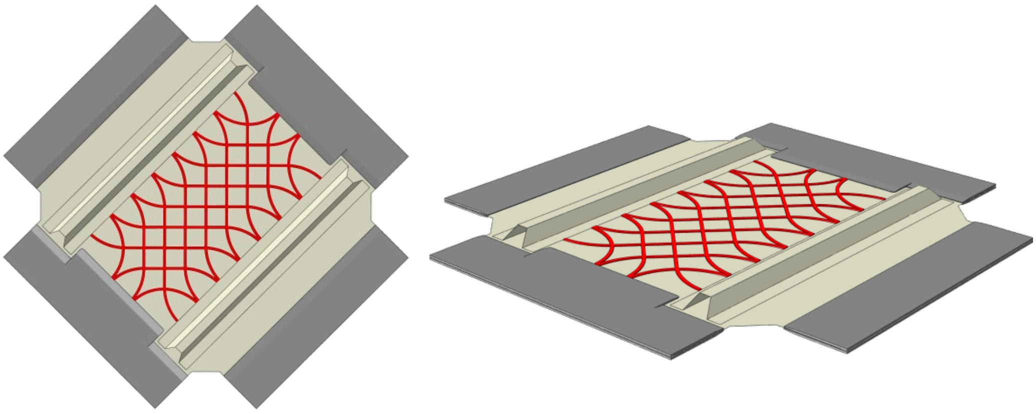


Figure 3: Grid stiffened shear panel design investigated in this study. The shear panel consists of a skin section and omega stiffeners indicated in light grey, aluminium tabs indicated in dark grey and the grid structure is placed in the wide central skin bay (pocket?) and is indicated in red

Table 1: Design values

Parameter	Value
Size total	760 [mm] (length/width shear panel)
Stringer cross section	125 mm^2
Stringer pitch	419 mm
Frame pitch	600 mm
Stiffener lam.: 9 PLY	[-45/45/0/0/90/0/0/45/-45]
Skin laminate: 10 PLY	[-45/45/90/0/-45/45/0/90/45/-45]
Material properties	Not public

By using grid stiffening as alternative stiffening means the aim is to reduce the weight of the entire forward fuselage section. The grid stiffening enables more freedom to the designer to specify each bay section in the fuselage according to the actual loads. For the shear test panel design the Reserve Factor (RF) for buckling was defined as such that 44 skin bay sections could be replaced using this technology. This would lead to a theoretical weight gain of 8.50 kg for the forward fuselage – around 1% weight saving.

For the detailed simulation model of the shear panel the skin part and stringers are modelled using conventional linear shell elements (S4R) in the ABAQUS finite element software [20]. For the grid stiffening section both beam elements and continuum shell elements are used.
The boundary conditions for the shear model are defined by a so-called picture frame used for the actual testing. This picture frame is hinged in its corners and can deform in a diamond shape and is modelled as steel beams by linear solid elements and connected by frictionless rotational joints in the corners. The bottom corner is loaded in upward direction and the top corner is fixed, similar to the physical test. The out-of -plane displacements are constrained to prevent rigid body modes of the test fixture.
In the following section the analysis procedure shown in Table 2 that was used for the shear panel design is discussed. This includes buckling, post-buckling, progressive damage and curing analyses.

Table 2: Overview of analyses performed on the grid stiffened structure. The respective methods are discussed in this section

Analysis type	Methods	Model
Buckling analysis (linear)	Lanczos	Reduced, shell, beam
Postbuckling Nonlinear analysis	Newton-Rhapson	Reduced, shell, beam
Progressive damage analysis	Cohesive surfaces	Shell, continuum shell
Damage tolerance (DT)	Cohesive surfaces	Shell, continuum shell
Manufacturing simulation (curing)	Thermal, mechanical HETVAL/USDFLD/UEXPAN User subroutines	Solid elements

The buckling analysis is performed using standard solutions to define the linear buckling bifurcation point [11]. This is performed using the Lanczos method and results are compared with test results. The post-buckling shear analysis is performed using standard solution for non-linear analysis. Results of both the linear buckling and non-linear post-buckling results are shown in the paragraph 3.1.

3.1 Progressive damage methods

Progressive damage simulation is used to determine whether separation between grid and skin occurs and when the skin fails. For the in-plane damage behaviour the Hashin 2D criterion is used and for the interface the surface cohesive formulation. Cohesive interaction is based on well-established traction separation laws that describe the relative displacement Δ of two connected surfaces and depending on the element stiffness determine the internal traction [20]. The interface damage modes that are assumed are Mode I (peel), and Mode II, III (shear), see Figure 4. The approach allows a linear softening of the interface when the damage is initiated.

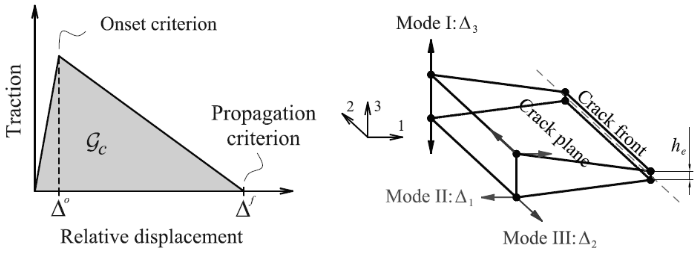


Figure 4: Traction separation graph depending on the relative displacement of the two connection surfaces and the mode discrimination in the cohesive model [8]

The inputs needed for the cohesive surfaces are the strength values for the damage initiation of the interface and the fracture toughness G_{Ic}, G_{IIc}, G_{IIIc} for the three modes. The interaction between the modes is determined using the Benzeggagh-Kenane (BK) mixed mode law shown in equation 1 [20].

$$G_c = G_{Ic} + (G_{IIc} - G_{Ic}) \frac{G_{II} + G_{III}}{G_I + G_{II} + G_{III}}^{\alpha} \qquad \text{Eq. 1}$$

Where G_c is the total mixed mode fracture energy, G_{Ic}, G_{IIc}, G_{IIIc} the critical fracture toughness energy and G_I, G_{II} and G_{III} the fracture toughness values during the simulation. The critical fracture toughness values are determined experimentally with Double Cantilever Beam (DCB) or End Notch Fracture (ENF) testing [13].

3.2 Curing analysis methods

The curing analysis has been newly developed using a sequential thermal and mechanical analysis to determine the residual stress and final deformation of the composite parts. In the curing simulation the rate of cure is an important factor to capture during the cure simulation. Because of the forming of linkages during curing of the resin, an exothermic reaction created. This can have an influence on the temperature in the mould. A limited description of the main formulations used in the user subroutines is given. The heat generated during the curing process is included in the formations as follows.

$$q(\alpha, T) = \rho \cdot Ht \cdot \frac{d\alpha}{dt} \cdot V_{mm} \qquad \text{Eq. 2}$$

Where ρ is the density, *Ht* is the enthalpy, $d\alpha/dt$ is the cure rate and V_{mm} is the matrix volume fraction. The cure rate is dependent on the actual cure state and the temperature (Kamal-Sourour [14]).

$$\frac{d\alpha}{dt} = f(\alpha, T) \qquad \text{Eq.3}$$

With $d\alpha/dt$ is the cure rate and α the cure state and T the temperature of the resin. This can be extended by using two pre-exponential coefficients using the Arrhenius rate expressions.

$$K = Aexp\left(-\frac{Ea}{RT}\right) \qquad \text{Eq.4}$$

Where A is pre-exponential coefficient, Ea is the activation energy, R is the universal gas constant, T is the absolute temperature. The formulation by Hubert [14] including auto-catalytic curing and diffusion factor is used in the presented curing model.

$$\frac{d\alpha}{dt} = K \frac{\alpha^m (1-\alpha)^n}{1 + \exp(C(\alpha - (\alpha c0 + \alpha ct \cdot T)))} \qquad \text{Eq.5}$$

Where α is cure state, m, n, C, $\alpha c0$ and αct are fitting factors that have to be extracted from experimental values using Differential Scanning Calorimetry (DSC) measurements. When the curing process takes place the glass transition temperature T_g increases. It is found that this can be described well with the diBenedetto equation [15]. This is however not actively used in the subroutine, only the T_g is calculated as output.

$$\frac{T_g - T_{g0}}{T_{g\infty} - T_{g0}} = \frac{\lambda\alpha}{1-(1-\lambda)\alpha} \qquad \text{Eq.6}$$

Here T_g is the glass transition temperature, T_{g0} and $T_{g\infty}$ are the glass transition temperatures of uncured and fully cured resin. Alpha is defined as the degree of cure and λ is a fit parameter between 0 and 1 for the specific resin.
In the mechanical model the stiffness of the material and in particular the resin is calculated using the Abaqus USDFLD user subroutine. It uses the formulation from Cisse [15] for the stiffness variation as factor of the temperature and the cure state. The strain in the model is calculated using the Abaqus UEXPAN user subroutine in which four strain factors are computed; the elastic strain, thermal strain, chemical shrinkage strain and creep strain. The thermal strain formulation is given in the following equation.

$$\Delta\varepsilon_{ij}^{therm} = \alpha(CTE)\Delta T \qquad \text{Eq. 7}$$

Here ε_{ij}^{therm} the thermal strain depending on the Coefficient of Thermal Expansion (CTE) denoted in this case with α and the temperature difference indicated with ΔT. Standard verification of this approach has been performed using L-shapes and existing composite structures. Validation has been performed by comparing the final spring in angle of the composite structures.

4 Numerical simulation results

In this section the numerical simulation are discussed using the previously shown formulations. This includes buckling and post-buckling analysis, progressive damage analysis and curing analysis. From the performed curing simulation result it has been decided to use a bonded interface between the skin and grid instead of co-curing. The risk of damage or separation of this interface during manufacturing was considered too large.

4.1 Manufacturing simulation

The model used for the nonlinear analysis is extended to perform manufacturing simulations. The curing process of the composite structure is simulated using finite element methods consisting of a sequential thermal and mechanical simulation. User subroutines are used to include the exothermal reaction of the resin and boundary conditions for the curing. The mechanical simulation uses subroutines to calculate the distortion and residual stress in the composite structure.
For the connection between skin and grid the interface is critical since there is a large stiffness change. This may cause large distortions/warping of the structure during curing and after the release. Different concepts for the manufacturing have to be used for the manufacturing simulation including combined skin and grid co-curing and separate grid curing. For the curing simulation the model has been modified to volume elements for the grid and skin structure instead of beam and shell elements. The temperature and stress distribution also needs to be calculated in the thickness directions. A result of the first cure simulation was that the combined skin and grid design caused significant deformations of the final product, and consequently also high residual stresses. In Figure 5 a result is shown of the co-cured shear panel where near the grid section, deformation can be observed in the order of 3.0 mm maximum out of plane.

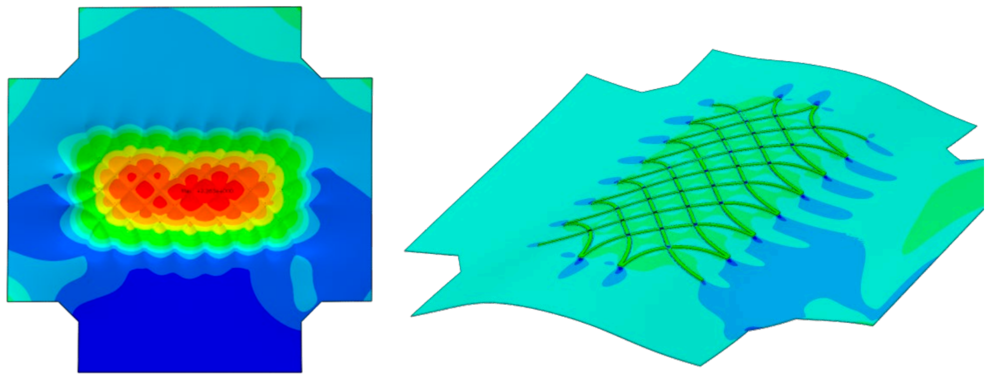


Figure 5: Results of the curing simulation with the out of plane deformation and maximum principal strains for the co-cured panel concept. Large displacements can be observed in the centre of the bay section

In the grid sections the residual stress was close to 20 MPa in compression and in the skin close to 100 MPa. With these curing results there was a large risk that the grid would separate during curing and render it obsolete. Therefore the save route was opted with manufacturing the skin and grid sections separately and bonding afterwards. Since both the skin and grid are symmetric lay-ups the curing distortions were minimal.

4.2 Buckling and post-buckling

Linear buckling analyses are executed for the grid stiffened shear panels as described in earlier sections. The Lanczos solver [20] is used for this analysis and the first two buckling modes are retrieved, see Figure 6. For the first mode a single dominant halve wave buckling can be observed over the entire bay section. The second mode is very different with a low difference in buckling load from the first mode. This indicates that either of these two modes can appear depending on imperfections in the panel.

The linear buckling values for the shear panel of mode 1 and mode 2 are: 43.6 N/mm and 44.6 N/mm, respectively. These values are a limit load requirement for the design of the shear panel. With the post-buckling analysis the loading up to ultimate load (150% limit load) is investigated, see Figure 6.

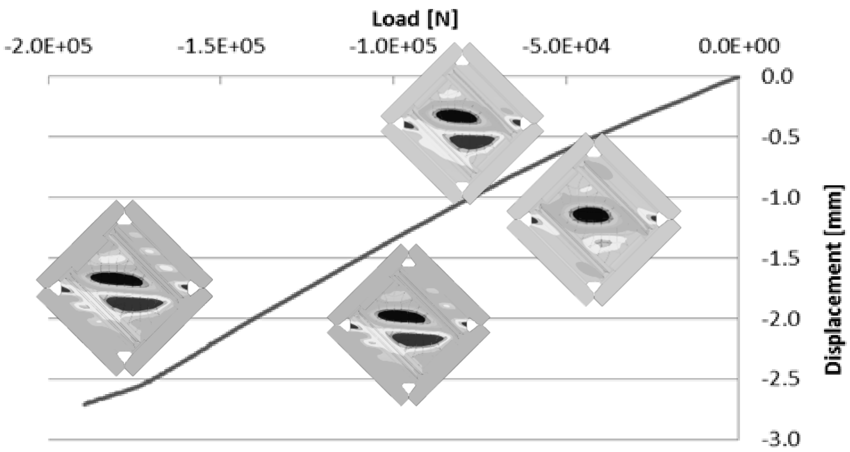


Figure 6: Post-buckling behaviour of the shear panel indicated for three nodes in the central area of the grid bay section. Initial instability occurs at ~65% of UL load

The shear panel with grid stiffening shows a stable post-buckling path with a mode transition from a single dominant halve wave to two dominant halve waves. With the out-of-plane displacement of the grid section over the diagonal this mode-jump can be observed, see Figure 7.

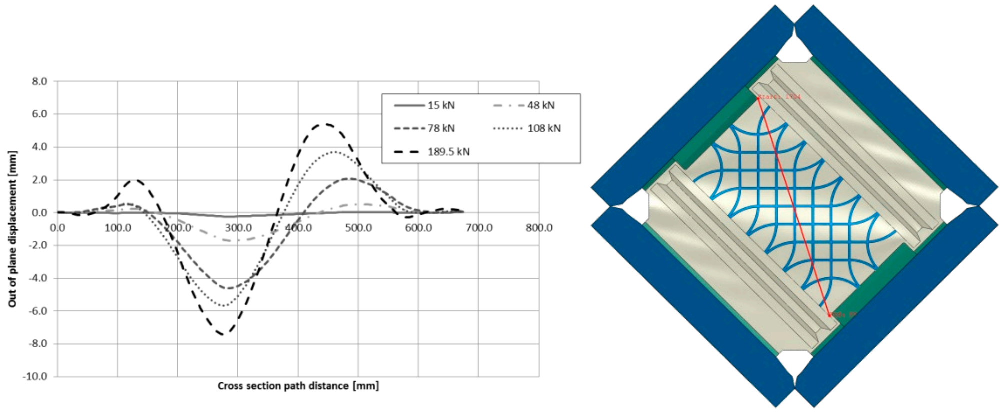


Figure 7: Out of plane displacement of the diagonal path of the shear panel grid section

Buckling analyses have been performed and will be compared later with the test results. To investigate possible damage or failure at ultimate load and above, the progressive damage methods as previously described are used and shown in the next section.

4.3 Progressive damage analyses

The progressive damage analyses are used to determine the strength and damage tolerance of the structure. This is an important measure for aircraft structures in case of damage occurring during the flight by for instance hail impacts. A nonlinear Newton Raphson solver technique is used for the analyses. The Hashin 2D composite failure method is used for the in-plane damage assessment. Interface elements have been placed between the skin and grid section and between skin and omega stringers. The material properties for the glued interface have been derived from test data however limited verification has occurred. Also the glue thickness which is an important factor in this is estimated. Therefore the results are mainly aimed at understanding the damage progressive pattern, and hence the results are more qualitative instead of quantitative.

Residual strength analyses have been performed to understand the sensitivity for certain damages. Impact damage is simulated with part of the mid-section grid separated from the skin. A second analysis is performed where the grid is separated from the skin and the grid itself also damaged at the intersection. These three situations are discussed in this section, Figure 8.

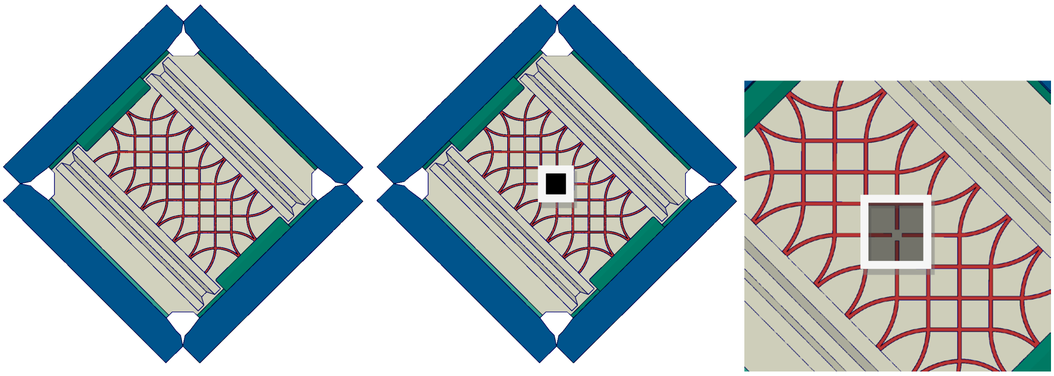


Figure 8: Simulation model for residual strength analysis. On the left the undamaged panel, in the middle, a debonding is included and on the right a debonding and cut of the grid is included

The interface damage pattern shows that damage is likely to occur at the edges of the omega stringer run-outs. This is an area in the panel where the load is transferred from the stringer to the skin and directly in the aluminium tabs. However initial failure of the panel structure is not resulting from this debonding, the material will fail in-plane at the corners. From the simulation results the interface damage will not initiate in the grid structure, see Figure 9.

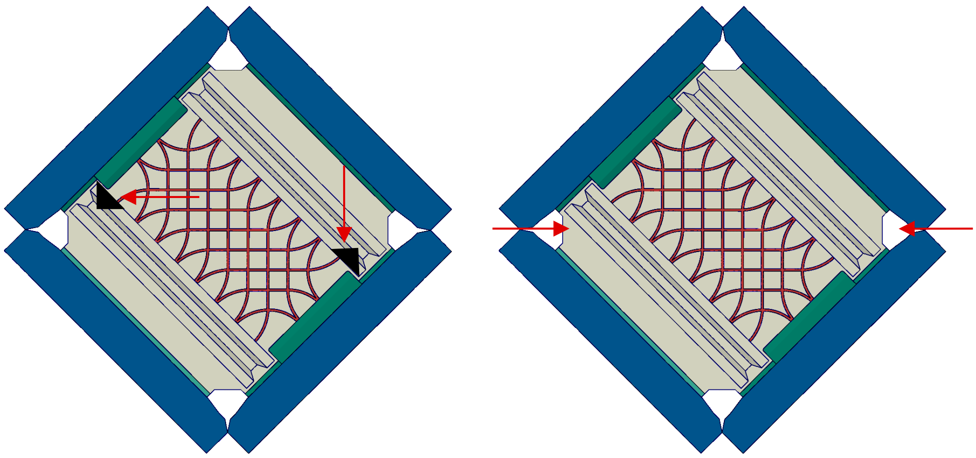


Figure 9: Damage indications with on the left the initial debonding of the stringer and on the right the initial failure of the composite material

Total debonding of these parts is not expected and first indication of damage in the panel will occur at a load of 128.3 kN which is around 2.5 times the limit load of the panel. This first failure will probably not lead to collapse of the panel since load can be diverted to other parts of the structure.

The residual strength analysis gave interesting results on the behaviour of the panel. The debonded area appears to have little influence on the overall stability and no local buckling of the skin occurs. This is mainly caused by the panel buckling mode which is concave in the region of the debonding, hence the grid is pressed against the skin. For the second residual strength analyses including cutting of the grid intersection the results are similar. On the global level a minimal influence of the damage on the behaviour can be observed, hence no reduction in initial failure load is expected. Although an increase in local strain in the skin is observed, this is still less than other critical parts in the panel. The damage is stable and minimal growth of the de-bonding is predicted, Figure 10.

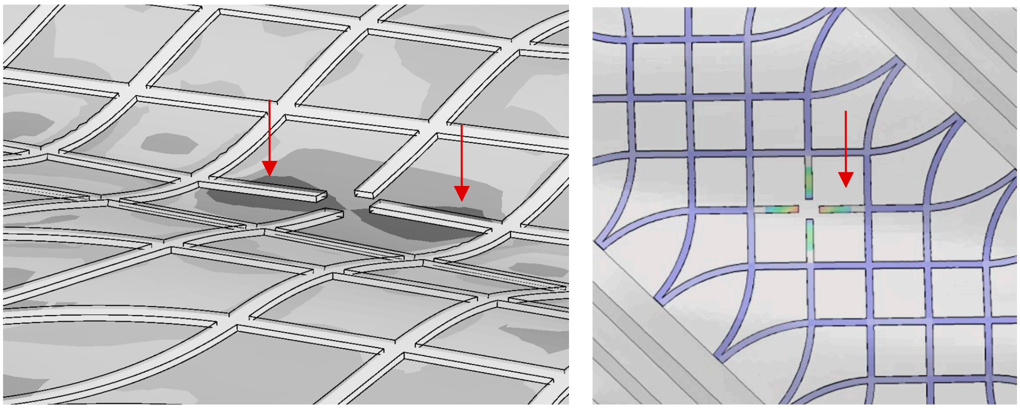


Figure 10: Residual strength analysis with debonded grid (partly) and removal of one grid cross section. Although an increase of strain in the skin is observed (left) the damage remains stable (right)

It has to be noted that the residual strength predictions can be sensitive to the actual buckling mode of the panel.

5 Manufacturing and testing

The composite shear panel with grid structure has been manufactured to investigate the process itself and also for later testing. The composite panel has been manufactured using pre-preg material and the skin and grids have been laid down by the NLR fibre placement machine. The skin and grid are manufactured and cured separately and bonded together afterwards. The connection of the grid with the omega stringer was designed using a spread out pattern to reduce the grid height. However the steering involved in the outer regions of the grid combined with the spread of the tows posed challenges to maintain a high quality. To ensure constant quality of the manufactured panel, additional c-scans were performed. The scan revealed a constant high quality of the skin laminate where in the grid outer areas some low quality laminates were observed. The manufactured grid, combined panel and c-scan image are shown in Figure 11.

Figure 11: Manufactured panel with the separate grid on the left and the combined panel including strain gauges, middle, and c-scan result on the right

Testing of the panels have been performed with the use of the before-mentioned picture frame setup and initially up to a load of 50 kN where initial buckling occurs, see Figure 12. The panels have been instrumented with strain gauges and the digital image correlation pattern for measuring the displacement and strain fields. Test results were very satisfactory with a good correlation of the initial buckling with the predicted results. The digital image correlation result for the out of plane displacement is shown in Figure 13.

Figure 12 (left) test setup of shear panel with grid. (right) test setup with shear panel and ARAMIS speckle pattern visible

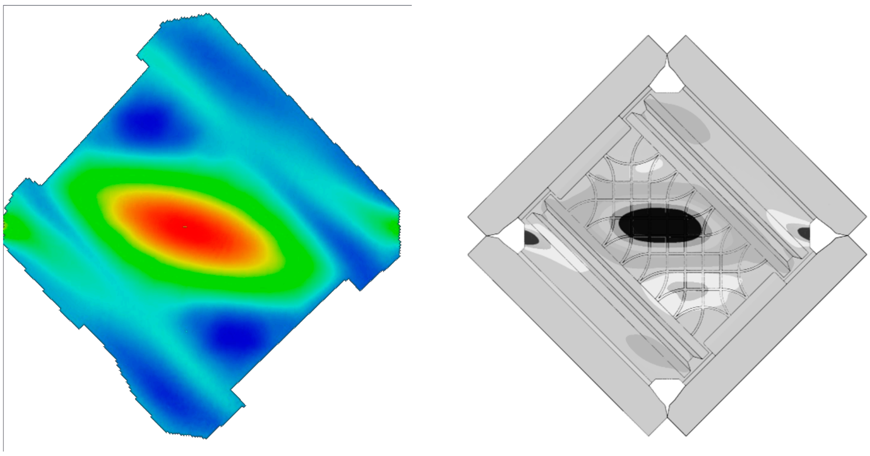


Figure 13: (left)Digital image correlation result from the test with a view on the outer side of the panel at 50 kN. The buckling mode corresponding to mode 1 of the finite element buckling analysis (right) can be observed

In the digital image correlation result a single dominant halve wave can be observed as first buckling mode. This corresponds to the buckling pattern observed in the simulations at low loads. Above this 50kN loading the buckling pattern will probably show a mode-jump. The overall stiffness of the panel and strain values correspond well with the predicted results, see Figure 14.

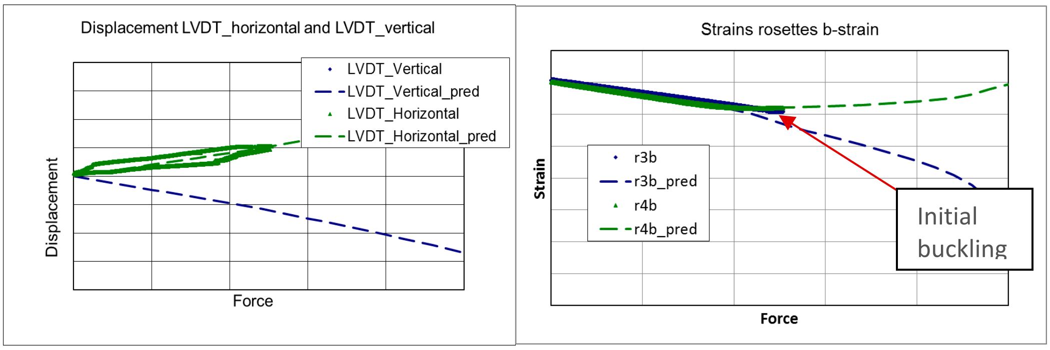


Figure 14: (left)Test results for displacement and strain (right) values of two rosettes. The vertical LVDT data was not included in the test

From the manufactured panels and the test results the confidence in the analysis and in grid structures is increased. The grid solution can provide a performance benefit in low loaded areas of the front fuselage section. However in this research some choices regarding manufacturability were made that has a profound influence on manufacturing time and cost.

6 Conclusions

This paper presents the investigation of several key aspects of a novel grid structure application. The grid structure posed challenges for manufacturing and in the end it was decided to perform bonding instead of curing due to the risk of distortion and interface damage during curing. The main conclusions can be summarized as follows.

- Simulation of grid structures as performed in this study is accurate for initial buckling and post-buckling predictions, which increases confidence for higher level simulations on fuselage level.
- Issues for manufacturing of co-cured grid stiffened composite structures still remain. These issues are mainly caused by the large residual stress on the interface of the grid and difference in thermal expansion coefficients.
- The chemical shrinkage of the grid worsens the interface issues mentioned because of the orthogonal alignment of the laminate (fully unidirectional).
- Progressive damage analyses give valuable insight in how the composite structure might fail. However for quantitative results a more extensive material testing programme needs to be performed for calibration of the model.
- Relatively low grid structures were successfully applied to 'reinforce' the skin and increase the buckling load effectively. A weight saving compared to the conventional structure can be realized in the low loaded areas of a fuselage.

Future work on this subject shall be focussed on investigations in a wider range of applications, such as wing or stabilizer components. Also the interface quality remains a topic of interest including producing grid structures with a large height (>25mm) without support during manufacturing. Open issues such as grid connection interfaces and repair still remain.

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