Fast Analysis Tools for Concurrent/Integrated Engineering of Composite Airframe Structures

R. Rolfes, J. Noack, F. Ruiz-Valdepenas, M. Taeschner, R. Zimmermann
DLR, Institute of Structural Mechanics, Braunschweig / GERMANY

1. Abstract

The terms *'Right First Time'* and *'First Right Time'* stand for the actual demands to the product development process, and they characterise Concurrent/Integrated Engineering as development procedure: The *right* design has to be obtained already at the *first time* when a prototype is manufactured, and the duration of the whole process has to be minimised, in order to enable the user to serve the market as *first* supplier at *right time*. Hence a basic prior condition for Concurrent/Integrated Engineering is the availability of procedures for structural analysis, which are as fast as required for the former conceptual phase and, simultaneously, as precise as for the former detailed phase of the design process. General Purpose Program Systems are ready for proof purposes, however, they are usually not quick enough for the sake of design; there is a strong need for new, efficient structural analysis methods.

For that purpose, at the Institute of Structural Mechanics of DLR procedures for fiber composite airframe structures are under development, which enable fast and precise analysis of damage behavior, 3D temperature distributions, 3D stress distribution and buckling behavior. Concepts and procedures for these analyses and their verifications by experiments will be presented.

2. Keywords

Concurrent Engineering, Composites, Finite Elements

3. Introduction

Typical aircraft design consists of three phases, which can be denoted as conceptual, pre-design and detailed design phase. Typically, in the conceptual phase some 30 different concepts are designed and corresponding manufacturing concepts are regarded. Usually, very rough analysis tools are applied which lead to an accuracy of the weight estimation of at most 92%. Based on such a data basis, the decision for one of the concepts can be risky, especially if there are equally ranked concepts. If the pre-design phase should reveal that the concepts chosen is not as promissing as expected, stepping back to the conceptual phase is necessary, but very costly.
A Concurrent/Integrated Engineering (CIE) process is proposed which comprises both, conceptual and pre-design phase. This leads to a decision making on a more soild data basis than presently. y use of new analysis tools, which are fast enough for conceptual design and at the same time accurate enough for pre-design purposes, time and costs can be reduced significantly as compared to state of the art analysis tools. The paper suggests methods for thermal and stress analysis, buckling as well as impact and residual strength analysis.

4. 3D Thermal analysis of composite and sandwich structures

Motivation

The main objectives of the thermal analysis is to control if the structure fulfills the thermal requirements and to supply the full three-dimensional temperature distribution as input for the thermo-mechanical analysis. In the case of composites and sandwich structures the layers have different thermal conductivities in different directions. Figure 1 shows different composites and a typical sandwich structure.

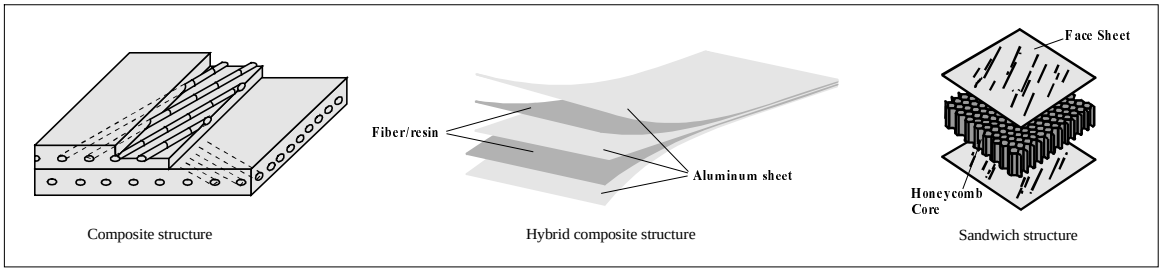


Figure 1: Examples of composite and sandwich structures

For the evaluation of the full 3D temperature distribution a three-dimensional finite element or finite difference model is necessary. This leads to high modelling and numerical effort, which is not acceptable within a design process. Besides, for thermo-mechanical calculations a two-dimensional model is sufficient, since most commercial finite element codes

provide two-dimensional finite elements for composite and sandwich structures. Finite elements which can calculate the full three-dimensional temperature distribution based on a two-dimensional model reduce the modelling and numerical effort drastically. For carbon fibre reinforced plastics (CFRP) *Rolfes* [1] has proposed a linear thermal lamination theory which is analogous to the first order shear deformation theory (FSDT). For local effects or transient problems the same author has suggested a quadratic thermal lamination theory [2].

3D thermal analysis based on 2D finite elements

CFRP, hybrid composites and sandwich structures can be idealised as layered structures (figure 2). For layers in which all modes of heat transfer (heat conduction, radiation and convection) occur (for example honeycomb cores) a thermal homogenisation is necessary. This homogenisation is not a specific requirement for two-dimensional finite elements, but is equally needed if a full three-dimensional finite element or finite difference model is applied.

For development of the two-dimensional thermal elements QUAD*LLT* and QUAD*QLT* two different temperature distributions in thickness direction were assumed. A linear layered theory (LLT) was first used by *Sipetov* [3] for steady state thermal problems. It assumes

$$T^k(x,y,z) = T_0^k(x,y) + z_k \cdot T_{0,z}^k(x,y)\,. \tag{1}$$

Using two heat transfer equilibrium conditions for the

- temperature at the layer interface,
- heat flux in transverse direction,

the number of functional degrees of freedom can be made independent from the number of layers. This theory was extended to transient problems by *Noack* and *Rolfes* [4]. For transient thermal problems and local heat loads a quadratic layered theory is assumed:

$$T^k(x,y,z) = T_0^k(x,y) + z_k \cdot T_{0,z}^k(x,y) + \frac{z_k^2}{2} T_{0,zz}^k(x,y)\,. \tag{2}$$

By use of a third heat transfer equilibrium condition for

- the change of the heat flux in transverse direction,

again the number of functional degrees of freedom can be made independent from the number of layers. Based on this theory the finite element QUAD*QLT* was developed.

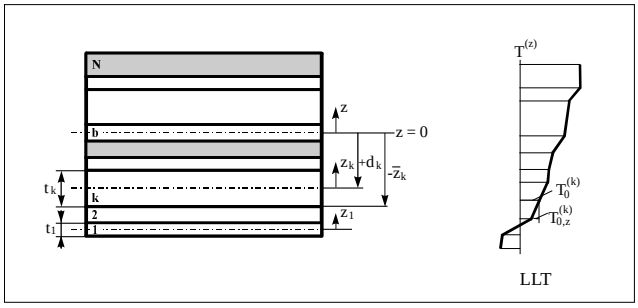


Figure 2: Layered design of composite and sandwich structures

Numerical example

A square plate with a local heat flux of $q = 100\ kW/m^2$ was considered. At the bottom of the plate a convection boundary condition with an ambient temperature of $T_\infty = 0\ °C$ and $\alpha_c = 30\ W/m^2K$ was applied. The geometry is shown in figure 3.

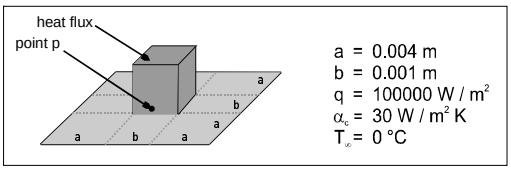


Figure 3: Example problem for the thermal analysis

Two different composites were analysed. The temperature distributions in transverse direction at point P are shown in figure 4. The results show a good agreement between 2D and 3D analysis. Numerical and modelling effort are drastically reduced by the new finite elements.

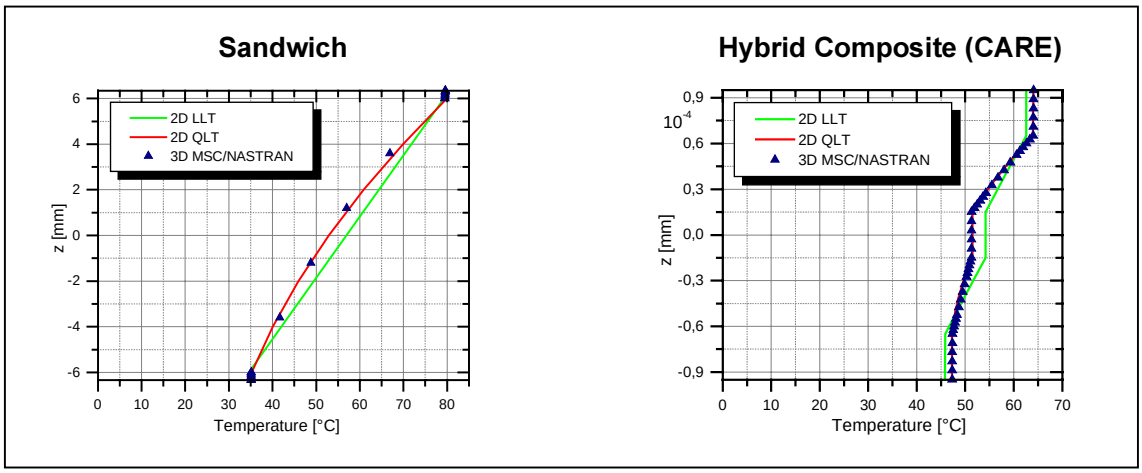


Figure 4: Comparison of 2D and 3D thermal analysis (transverse temperature distribution at point P)

5. 3D stress analysis of composite structures

For many years the failure of composite laminates has been predicted by so called quadratic interaction criteria for single unidirectional layers, which can be thought of as siblings of the *von-Mises* yield stress criterion extended to anisotropy. Neither do they provide information about *how* the laminate fails nor do they account for transverse stresses. Though lightweight structures are typically thin-walled, laminated composites show reduced strengths in transverse direction at the same time, caused by the dominating matrix material. To circumvent this situation, new physically motivated failure criteria have been proposed e.g. by *Hashin* [5], *Puck* [6] and *Cuntze* et al. [7], that differentiate between specific modes of failure. Those criteria do account for transverse stresses as well, therefore require the full 3D stress tensor to provide more accurate failure prediction.

Since analytical solutions are usually limited to special configurations, general finite element codes are employed to perform real world analyses. Obviously, it is possible to discretize any composite structure using 3D solid elements and gain the 3D stress state, but this approach is highly impractical due to high model handling and calculation costs. As an alternative a fast, cost efficient method to determine the full 3D stress state has been proposed by *Rolfes* et al. [8,9,10], lately. It is applicable to laminated composites under both temperature and mechanical loads and especially suited for early design cycles.

The method comprises two stages: At first a FEA deformation analysis is performed, using standard 2D finite elements based on FSDT. Such elements are available in virtually all general purpose FEA solvers. Displacements and laminate strains on the reference surface are used to calculate layerwise membrane stresses and transverse shear resultants as well as their first derivatives. The postprocessor TRAVEST uses this as input for the second step. The full three-dimensional conditions of equilibrium are to be integrated over the thickness utilizing the material law plus two simplifying assumptions:

- The influence of membrane normal stress derivatives on transverse shear is neglected.
- Two cylindrical bending modes are assumed to describe the deformation behaviour.

These simplifications allow for reduction of regularity requirements on the shape functions of the finite elements. With biquadratic trial functions the complete stress recovery can be performed on element level, avoiding global interpolations. The integration process needs to be performed only once per laminate definition.
Extensive tests using three dimensional elasticity or finite element as reference have shown good, sometimes excellent coincidence for transverse stresses despite the simplifying assumptions even for moderate thick plates and shells.

Two examples may be presented here: A simply supported cross-ply plate under doubly sinusoidally distributed temperature gradient[1], constant through thickness and a cylindrical, long panel under sinusoidal pressure load [11]. In the plate problem two different slenderness ratios were investigated, l/h=10 and l/h=20. Both examples show a very good agreement of both transverse shear and transverse normal stresses (figure 5).

The extended two-dimensional method has proven to provide an accurate estimation of all transverse stresses on element level while reducing modeling and calculation cost compared to three-dimensional FEA models considerably. Currently the method is being generalized to doubly curved shells and a reference implementation in the MSC/PATRAN - Laminate Modeler environment is work in progress. A test release will be available to the public in due term.

[1] Reference solution kindly provided by Prof. A.K. Noor of NASA Langley research center, Hampton VA, USA.

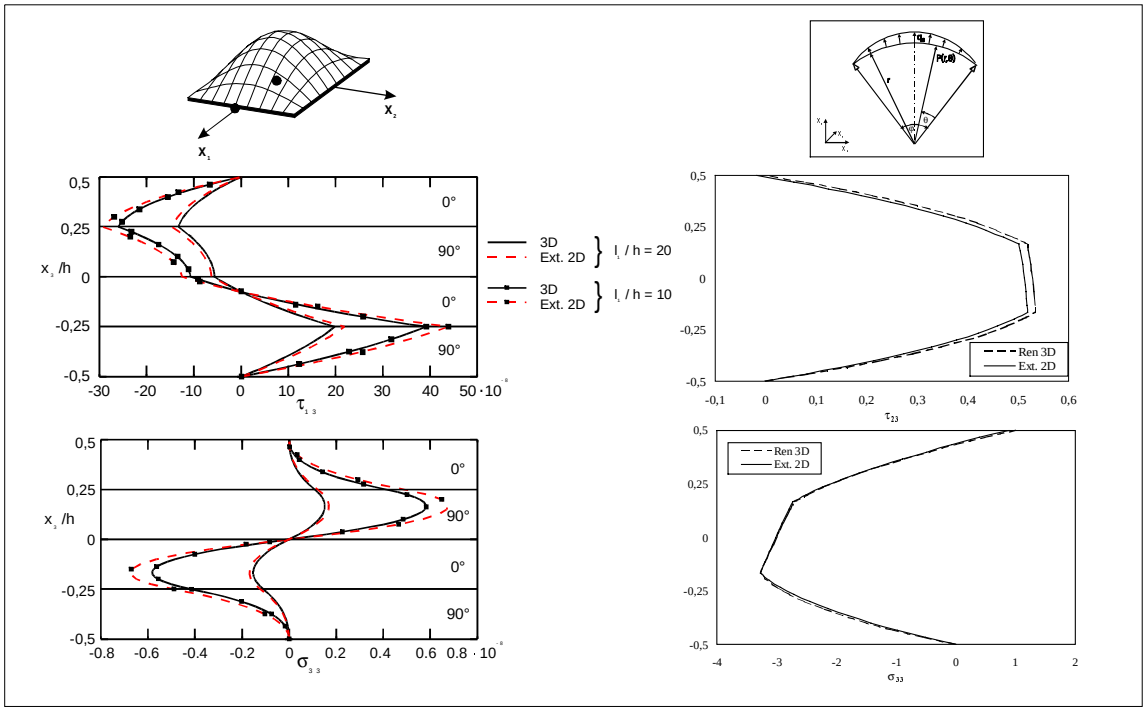


Figure 5: Transverse stresses in plate and shell under thermal and static loading, respectively

6. Stability

In order to substantially reduce weight of aircraft without prejudice to costs and safety, increased use of advanced fiber composites for primary structures and exploitation of all structural reserves are challenges emphasized by aircraft industries all over the world. In this respect an actual objective is to design fiber composite fuselage structures for postbuckling under ultimate load. Figure 6 explains the objective; it shows the measured load shortening diagram of an axially compressed, stringer stiffened, curved CFRP panel (the stringers are located at the rear of the panels), [12]. It clearly can be seen, that with this panel a load increase of 20% beyond the knee is possible before final fracture. The knee is due to skin buckling at 100 kN, which indicates the begin of the postbuckling regime. When applied to real fuselage structures, going into the postbuckling regime must be possible repeatedly without detremental influence on the structural properties. In this context, in particular skin stringer connections must remain uninjured.

Use of advanced materials, new manufacturing procedures, improved design philosophies and other innovative actions taken to reduce weight may impact safety, even with aspects for which - based on actual experience - no effect could be expected. One of such aspects is buckling due to landing impact. The dynamics of landing load may interact with the dynamics of the buckling process itself in such a way, that buckling load reduction in comparison with predictions based on quasi-static assumptions, may occur. Figure 7 visualizes the phenomenon; it shows an assumed normalized buckling load versus normalized impact period. With very short period the buckling load is higher than that for quasi-static buckling, and with long period it coincides with this one. However, between them there is an area, for which the buckling load is substantially smaller than the quasi-static one; it is called the critical domain, because it presently is not addressed either with computational or with experimental proof of aircraft structures.

Although apperently structural failure due to this phenomenon has not been reported in the open literature, this effect has to be considered in any case with the above mentioned innovative actions aiming at weight reduction. An example is given in Figure 8 on buckling of an axially compressed advanced composite cylindrical shell with radius R = 250 mm and length L = 510 mm, [13]. It consists out of 4 CFRP layers of 0.125 mm thickness each, from outside to inside oriented at +41°, -41°, +24°, -24° to the cylinder axis. The shell is dynamically loaded with a rectangular time history. The dynamic buckling loads given are computed by use of ABAQUS Explicit for an imperfection shape as that belonging to linear bifurcation, and an amplitude of 25% of the laminate thickness. The results clearly show, that for very short time duration the dynamic buckling load is higher than the quasi-static one, whereas with with longer duration it comes down to less than 55% of it. These results need corroberation by further computations and in particular by experiments. Therefore the buckling test facility of DLR recently has been extended to the ability of dynamic loading. It should be mentioned, that with Figure 7, in opposite to Figure 8, a linear increase of loading time is assumed, which is more realistic than the rectangular one used with the first computations.

For both topics - and other ones in the field of buckling, which are not mentioned here - improved simulation tools and new design procedures for stiffened fiber composite panels are needed. Among others, the influence of laminate set-up has to be modelled carefully, because it strongly influences buckling and postbuckling behaviour as well as the sensitivity to imperfections, [14, 15]. For instance, it has been shown by computations and tests, that variation of the laminate set-up of axially compressed cylindrical CFRP shells influences the buckling load by a factor of up to about three , and that even reversing the stacking may influence it by a factor of two. Choice of laminate set-up substantially effects the quality of the design, and it must be done preferably by systematic optimization.

As yet the calculations of postbuckling and of buckling due to dynamic loading are extremely time consuming. Hence, fast and reliable procedures ready for industrial application have to be developed, which reduce design and analysis time by an order of magnitude and thus constitute an indispensable contribution to multidisciplinary optimization and concurrent engineering. This remarkably will reduce weight, improve safety, and minimize response-to-market time.

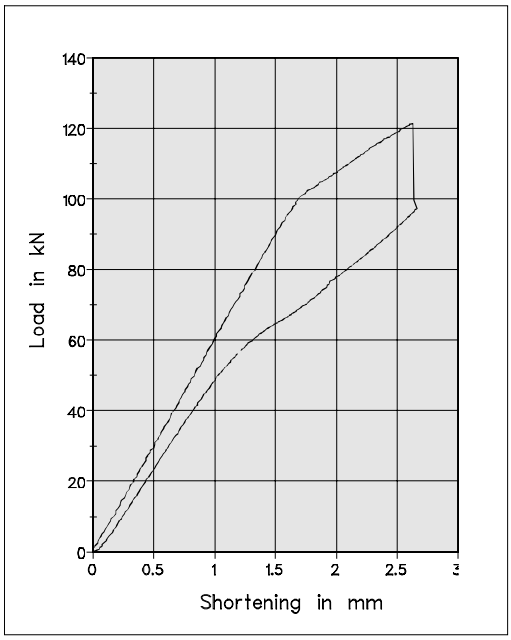


Figure 6: Measured load-shortening diagram of stiffened, curved CFRP panel

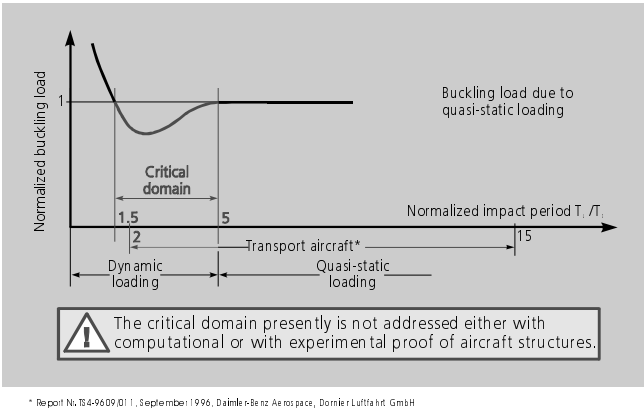
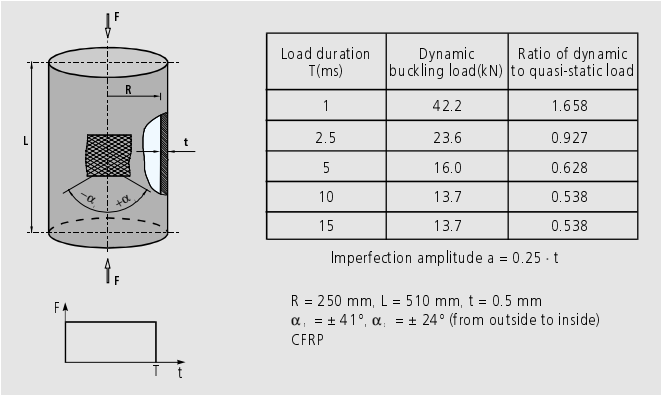


Figure 7: Buckling due to dynamic loading: Landing impact



Load duration T(ms)	Dynamic buckling load(kN)	Ratio of dynamic to quasi-static load
1	42.2	1.658
2.5	23.6	0.927
5	16.0	0.628
10	13.7	0.538
15	13.7	0.538

Imperfection amplitude a = 0.25 · t

Figure 8: CFRP shell subjected to impulsive axial compression

7. Impact and residual strength of composite structures

In order to be able to efficiently design composite structures with increased damage tolerance, it is essential to develop fast analysis tools which simulate the behaviour of the structure under service load conditions. In the context of concurrent integrated engineering, these simulation tools approach two objectives: to make available fast predictive methods for the preliminary design and to minimise the amount of tests (from coupon tests to real-size parts) needed in the verification of the structure.

One of the most important challenges in composite structures is the estimation of residual strength of damaged structures. It is essential to have the possibility to simulate the damage in a composite laminated structure produced by low-energy impact, which is one of the more limiting factors in real structures, and estimate the residual strength of the impacted structure. Both capabilities have been implemented in the computer programme CDTAC (Computer Damage Tolerance Analysis Code), developed in the Institute of Structural Mechanics of DLR [16].

The mechanisms of impact damage initiation and propagation are very complicated and have not yet been fully understood. Extensive experimental studies have, however, identified some important characteristics of impact damage patterns, especially concerning delaminations. The models included in CDTAC [17] are based on these basic characteristics of impact delamination identified by the experimental studies and focus on the qualitative description of geometric features (shape and distribution through the thickness) of impact delamination. Unlike previously proposed models, CDTAC does not require any parameters depending on impact tests.

The impact response is simulated using Herztian indentation law to calculate the contact force between impactor and laminate. The impactor is considered as a rigid mass which transfers a point load onto the laminate. A finite element analysis is made, where the laminate is modelled with plate elements based on Mindlin's plate theory to consider transverse shear effects. The dynamic equations of motion are then solved by direct integration methods. Thus, displacements of each node are determined, strain and stresses can be calculated over the laminate at each time-step, and failure criteria are applied to assess the initiation of damage, which allows to monitor the evolution of damage during the impact event. Three different failure criteria are respectively considered for fibre breakage, matrix cracking and delamination.

Figure 9 shows the impact transient response obtained by simulation with CDTAC compared with experimental measures. Figure 10 shows the distribution of impact damage simulated with CDTAC.

In order to record and characterise the damage information as much accurate as possible in the course of the analysis, the Damage Data Structure (DDS) has been implemented. The DDS is used in both parts of the programme, impact and compression after impact analyses. This allows the generalisation of the methods for different types of failures and arbitrary distribution of the damage in the laminate.

Extensive experimental studies have shown that the failure of impact-damaged laminates under compression is generally governed by the local buckling of the delaminated areas. Therefore, the Compression After Impact approach [18] is based on sublaminate buckling analysis, in which the delaminations obtained by impact simulation are assimilated to equivalent ellipses. Then, the stiffness of the buckled areas is modified by a stiffness reduction factor and a soft-inclusion is constructed in the damaged zones of the laminate [19]. With linear static finite elements analysis, the stress distribution is calculated and the Damage Influence failure criterion [20] is applied to obtain the compressive failure load of the laminate. Figure 11 shows results of compressive residual strength obtain by CDTAC compared with experiments.

The programme CDTAC, initially developed for simple composite laminates has recently been extended to a new version NCDTAC for the analysis of stringer-stiffened panels, based on the same ideas explained above, which means a further step towards the simulation of impact and CAI in real structures [21].

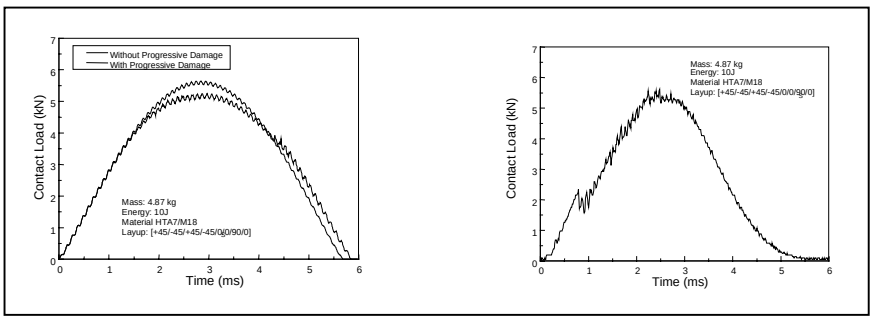


Figure 9: Impact transient response

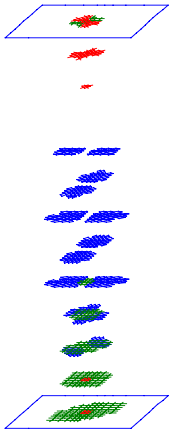


Figure 10: Impact damage simulation

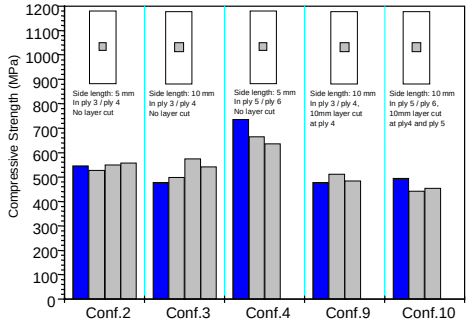


Figure 11: Compressive residual strength

8. References

[1] Rolfes, R.: Efficient Thermal Analysis of Anisotropic Composite Plates Using New Finite Elements, in: Füller, J.; Grüninger, G.; Schulte, K.; Bunsell, A.R.; Massiah, A.: Developments in the Science and Technology of Composite Materials, Proceedings of the Fourth European Conference on Composite Materials (ECCM4), Stuttgart, pp. 743-748 (1990)

[2] Rolfes, R.: Higher Order Theory and Finite Element for Heat Conduction in Composites, in: Lewis, R.W.; Chin, J.H.; Homsy, G.M.: Numerical Methods in Thermal Problems, Vol.VII, Proceedings of the Seventh International Conference, Stanford, U.S.A., pp. 880-889 (1991).

[3] Sipetov, V.S; Karpilovskii, V.S; Demchuk, O.N.: Application of the Finite Element Method to Solve the Stationary Heat Conduction Problem of Piecewise Inhomogeneous Systems. In: J Eng Phys 55, Nr. 6, S. 1439-1443, June 1989.

[4] Noack, J.; Rolfes, R.: Efficient thermal analysis of hot structures using new finite elements. Proceedings European Conference on Spacecraft Structure, Materials and Mechanical Testing 1998, Braunschweig, Germany, pp 117-12

[5] Hashin, Z.: Failure Criteria for Unidirectional Fiber Composites. In J. Appl. Mech., 1980, Nr. 47, pp.329-334.

[6] Puck, A.: Festigkeitsanalyse von Faser-Matrix-Laminaten. Carl Hanser Verlag, 1996.

[7] Cuntze, R.; Deska, R.; Szelinski, B.; Jeltsch-Fricker, R.; Meckbach, S.; Huybrechts, D.; Kopp, J.; Kroll, L.; Gollwitzer, S.; Rackwitz, R.: Neue Bruchkriterien und Festigkeitsnachweise für unidirektionalen Faserkunststoffverbund unter mehrachsiger Beanspruchung. VDI Verlag., 1997, Fortschrittsberichte VDI, Reihe 5, Nr. 506.

[8] Rolfes, R.; Rohwer, K.: Improved Transverse Shear Stresses in Composite Finite Elements Based on First Order Shear Deformation Theory. Int. J. Num. Meth. in Eng. (40) 51-60, 1997.

[9] Rolfes, R.; Rohwer, K.; Ballerstaedt, M.: Efficient linear transverse normal stress analysis of layered composite plates. Computers & Structures 6(68) p.643-652, 1998.

[10] Rolfes, R.; Noor, A.K.; Sparr, H.: Evaluation of Transverse Stresses in Composite Plates Based on First-Order Shear Deformation Theory. Computer Methods in Applied Mechanics and Engineering 167, pp.355-368, 1998.

[11] Ren, J.G.: Exact Solutions for Laminated Cylindrical Shells in Cylindrical Bending. Composite Science and Technology 29, pp. 169-187, 1987.

[12] Geier, B., Zimmermann, R.: *Fracture Mechanics of Composites.* ESA/ESTEC Contract No. 10.226/92/NL/PP(SC)S, Final Report, September 1996.

[13] Bisagni, C., Zimmermann, R.: *Buckling of Axially Compressed Fiber Composite Cylindrical Shells due to Impulsive Loading.* Conference on Spacecraft Structures, Materials & Mechanical Testing, Braunschweig, Germany, 04-06 November 1998, Proceedings (ESA SP-428, February 1999), p. 557-562.

[14] Geier, B., Klein, H., Zimmermann, R.: *Buckling Tests with Axially Compressed Unstiffened Cylindrical Shells Made from CFRP*. In: Buckling of Shell Structures, on Land, in the Sea and in the Air, J.F. Julien (ed.), Elsevier Science Publishing Co., Inc., London and New York, 1991, p. 498-507.

[15] Zimmermann, R.: *Quick Optimum Design of Axially Compressed, Thin-Walled FRP Cylindrical Shells*. 5th AIAA/NASA/USAF/ISSMO Symposium on Multidisciplinary Analysis and Optimization, Panama City, Florida, Sept. 7-9, 1994, AIAA Paper 94-4366-CP, p.1046-1056.

[16] Tang, X.; Shen, Z.; Chen, P.; Gädke, M.: Methodology for residual strength of damaged laminated composites. Proceedings of the 38th AIAA/ASME/AHS/ASC Structures, Structural Dynamics and Materials Conference, CP-AIAA-97-1220, 1997.

[17] Tang, X.; Gädke, M.; Weber, R.: Impact damage simulation and compression after impact (CAI) of laminated composites. DLR Internal Report IB 131-97/37, 1997.

[18] Tang, X.; Gädke, M.: Study of strength of carbon-fiber composite laminates with rectangular inserts and ply-cuts. DLR Internal Report IB 131-97/36, 1997.
[19] Dost, E. F.; Ilcewicz, L. B.; Gosse, J. H.: Sublaminate stability based modelling of impact-damaged composite laminates. Proceedings of the 3rd Technical Conference of American Society of Composites, Sept. 1988.
[20] Tang, X. D.; Shen, Z; Chen, P.H.; Yang, S. C.: Approach to residual strength of laminated composites with damage. ASRI Technical Report, Q/623S-9503-25, 1995.
[21] Chai, Y.; Gädke, M.: Impact damage simulation and compression after impact of composite stiffened-panels. DLR Internal Report IB 131-99/20, 1999.