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Damage propagation in composite structural elements - Analysis and experiments on structures

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ABSTRACT
An overview is given of experiments and analyses, being performed at the structures level within the GARTEUR (Group of Aeronautical Research in EURope) Action Group AG16: "Damage Propagation in Composite Structural Elements".
Experiments have been carried out on sandwich structure, stiffened flat and curved panels and stiffened cylinders, and on derivative simplified specimens.
The specimens were provided with impact damage or artificial defects. New test methods are being developed and detailed measurements are being carried out to determine damage growth and failure mechanisms. The work should result in design guidelines and analysis methods applicable to the type of structures that have been investigated.

Summary

An overview is given of experiments and analyses, being performed at the structures level within the GARTEUR (Group of Aeronautical Research in EURope) Action Group AG16: "Damage Propagation in Composite Structural Elements".
Experiments have been carried out on sandwich structure, stiffened flat and curved panels and stiffened cylinders, and on derivative simplified specimens.
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1 Introduction

Damage propagation in composite structural elements has been the subject of many research efforts, and is still not fully understood. For a damage tolerant design, the performance of a structural element must be adequate, even in the presence of damage, which must be assumed to have been introduced during production or during service. As damage propagation under load may lead to structural degradation, understanding this phenomenon is essential for an efficient design process. Without this understanding, the design process of damage tolerant composite structures would be reduced to a trial and error process.

In order to make more progress in this respect, the efforts of several partners have been combined within the framework of GARTEUR (Group for Aeronautical Research and Technology in EURope), and an action group has been established (AG 16: "DAMAGE PROPAGATION IN COMPOSITE STRUCTURAL ELEMENTS"). This action group has started its work in 1994, and was scheduled to be active during two years. Contributions to this action group are made by representatives of industry and research institutes of the participating countries: Aerospatiale and ONERA from France, DB-Aerospace/Airbus, Eurocopter Germany and DLR (two different institutes) from Germany, CIRA from Italy, NLR from the Netherlands, Saab-Scania and FFA from Sweden, and British Aerospace and DRA from the United Kingdom.

The joint effort to improve the understanding of damage propagation in composite (carbon-epoxy) structural elements is based on twelve individual studies, carried out by the participants with their own funding. These studies all involve extensive experimentation, while numerical analysis is involved in most studies. The major task of the action group is to satisfy the objectives that were formulated at the start of the activity. Hereto, discussions have been taking place during meetings twice a year, at which interaction of the individual research efforts is pursued. The findings will be presented in several reports. To structure this process, the work was divided into two areas: one group of participants is performing fracture mechanics type experiments, such as DCB and ENF, and relating the results to damage growth experiments at the coupon level (the COUPONS group), the other group is performing experiments on structural elements, relating the results to damage growth results obtained with more simple specimens, (the STRUCTURES group).

The present paper describes the work carried out so far by the members of the STRUCTURES group, the authors of this paper. The first part of the paper describes the background of the subject: design for damage tolerance, based on insight in damage initiation and propagation in composite structures, and lists the objectives of the programme. The second part of the paper gives an overview of the six individual contributions, focusing on the particular background, objectives, applicability and approach of each. As the work has not been finished yet, the discussion of the results is limited to some interesting highlights.

2 Background

The design of structures for aerospace applications is subject to many constraints: these structures should be light weight, efficient to produce, affordable to maintain, safe, acceptable to the environment, etc. Focusing on the light weight criterion, structural designs can be made light weight by choosing light materials and by selecting efficient structural configurations.
Typical light weight materials are fiber-reinforced composites (e.g. carbon/epoxy), while typical structural configurations with high efficiency are shell structures. Many new aerospace structures are therefore designed as fiber-reinforced composite shell structures. Designing such structures for high performance is "state-of-the art", but designing for damage tolerance is an area that still needs a significant amount of work. Other major challenges for the development of highly loaded composite structures for application in civil aircraft are to make their production and maintenance affordable.

Materials. High performance fiber-reinforced composite materials are made of continuous (long), strong and stiff fibers, which are aligned in predetermined directions, and are embedded in a relatively weak and soft matrix. Therefore, such materials distinguish themselves from conventional materials by their anisotropy and their non-homogeneity. Anisotropy dominates the stiffness properties, while non-homogeneity has a major influence on the strength properties. Nowadays, most stiffness problems are well understood, but strength prediction is still far from being mastered, especially when the presence of damage must be considered. Aerospace applications require damage tolerant structural designs, designs which perform adequate, even when certain, well defined damages are present in the structure. Such damages may have been induced in the structure during production or during service.

Structures. Shell structures are efficient structures because the loads are carried in the plane of the structural elements. To carry in-plane loading, thin plates are usually sufficient. The minimum plate thickness is often governed by buckling requirements, a stiffness constraint, rather than by strength requirements. Buckling may occur when the in-plane loading is a compression load or a shearing load. To improve the buckling characteristics of a thin plate, stiffening concepts are used, which is a more efficient solution than increasing the plate thickness. Common stiffening concepts are skin-stiffener configurations or honeycomb sandwich configurations. Other types of loading of shell structures may be moderate transverse pressures or pressure differences.

Fabrication. Most common fabrication methods for high performance composite plates with continuous fibers are constrained to produce laminated structures.
Laminates consist of several layers with uni-directional fibers, while each layer may have a

different fiber orientation. The vulnerability of this concept is the lack of strength in the transverse direction of the plate, because of the weak interfaces between the layers. Certain new technologies such as stitching or resin transfer moulding with 3-D fiber structures may solve this problem, but many structures will still be made with the traditional laminating techniques.

Damage tolerance. One of the major failure modes that has to be considered when designing high performance fiber-reinforced, laminated composite structures for damage tolerance is the result of the combined aspects described above. The weak interfaces between layers with uni-directional fibers are vulnerable, and may fail due to impact damage. The resulting delaminations create several sublaminates, which in conjunction are less stiff than the original plate, reducing stability when the plate is loaded by in-plane compression or shearing loads. Moreover, delaminations may grow under service loads, further weakening the structure, so it will no longer be able to carry the ultimate design load. The buckling of sublaminates, formed by delaminations in the plate, is one of the mechanisms resulting in the onset of delamination growth and further propagation. Other weak interfaces where delaminations can initiate and propagate are adhesive layers connecting a skin plate to a stiffener or to a honeycomb core.

Design technology. Today, damage tolerant structures are often designed with empirical design guidelines which are based on experience resulting from case studies and earlier applications. The design process, by nature a cyclic process, depends on extensive testing, which is expensive and time consuming, hence, should be optimized to limit the number and scope of the design cycles. Therefore, there is a need for (i) improved design guidelines, to find solutions in fewer cycles, (ii) efficient design analysis methods, to evaluate the performance of a particular design more accurately and (iii) simple but relevant test methods, to reduce the time and cost of a design cycle. The development of these three design tools must be based on a thorough understanding of the relevant failure mechanisms, for which well focused experimental programmes must be carried out. It can not easily be avoided, however, that these programmes are related to the particular structural configuration considered, an approach followed by the STRUCTURES Group. This is different from the approach followed by the COUPONS group, which is more generic: aimed at developing a method to predict (the onset of) delamination growth in a part of any type of structure. This method is based on fracture mechanics type experiments, and on fracture mechanics type analyses. The complexity of such a method, however, makes it probably more suitable for application in the final design validation phase, than for the initial design phase.

3 Objectives

The objectives of the work performed by the STRUCTURES group are in line with the overall objectives of the action group, and can be summarized as follows:

- improve the understanding of initiation and growth of delaminations by careful experimentation,
- develop and validate efficient design analysis methods,
- develop simple test methods using simple specimens, and determine how damage growth in these specimens relates to damage growth in structural elements,
- develop design guidelines, establishing important parameters that influence damage propagation, (which may be related to structural features).

These objectives are addressed in the following presentation of the individual contributions.

4 Sandwich panels

Composite sandwich panels are often used as fuselage shells of helicopters. A design methodology for such panels requires that the strength reduction due to impact damage can be predicted, both in case of static and fatigue loading. Load cases are usually multi-axial, i.e., combined in-plane compression and shear. Eurocopter Deutschland (ECD) has undertaken a research effort to determine the damage growth onset in case of static and fatigue loading. An analysis method is being developed to extend a simple failure criterion, originally developed for static compression, to the multi-axial static load case. Subsequently, data is being gathered in order to establish design guidelines pertaining to damage growth in case of shear under low cycle fatigue loading.

The investigation focused on a single configuration, consisting of [0/90,±45,0/90,±45,0/90] skins made of 913C/T300 woven fabric on a 15 mm Nomex honeycomb core (48 kg/mm^3). First, static compression tests were carried out on 150 x 100 mm specimens with impact damage, mounted in a test fixture as shown in figure 1, with clamped loaded edges and simply supported lateral edges. Secondly, static and fatigue tests in shear were carried out on square 500 x 500 mm specimens with flanges, clamped in a picture frame as shown in figure 2. Impact damages were applied with a 25mm diameter hemispheric tup at energy levels of 5J to 30J. Loads, strains, and displacements were recorded, while the damage was assessed with C-scans.

The analytical approach of ECD is twofold. First, a semi-empirical model is applied which provides the residual compressive strength of the panel as a function of damage size. The key assumption underlying this approach is that impact damage causes a stiffness reduction in the impacted area of the skin, resulting in stress concentrations in the surrounding undamaged material when loaded. These stress concentrations then initiate premature failure, which can either be a skin rupture or loss of local stability in the vicinity of the impact area. The results of the compression tests are required as input for the failure criterion used. Secondly, a global failure criterion applicable to multi-axial loaded, impact damaged panels is being evaluated, using the results of the static shear tests. This criterion is based on a knock-down factor derived from the compressive strength reduction. It is surmised that differences in the mechanical performance of impact damaged sandwich panels are mainly caused by a stress redistribution in the sandwich skins.

Composite sandwich panels are also often used in light structures of civil aircraft, such as rudder, dorsal-fin, flaps and floor panels. Face sheets may be as thin as one ply, which makes the quality of the skin-to-core bond-line even more important. Critical loading conditions are in-plane shear and compression loading, combined with differential pressure loadings, statically or in fatigue. Daimler-Benz Aerospace Airbus (DBA) has undertaken a research effort to investigate the influence of the quality of the bond-line as influenced by variations of material, production methods, and service life. A design methodology for damage tolerant sandwich panels requires that the onset and growth of face-sheet debonding is understood, and that improved design guidelines are established. Although fatigue tests using internal pressure loading on circular specimens were shown to be quite rigorous, there is a strong need for a more simple test method, for which a modified drum-peel test has been developed.

A test programme has been carried out by DBA to determine the onset of face-sheet debonding under static and fatigue normal peel loading, depending on core geometry and bond-line characteristics (formation of adhesive meniscus). For this purpose a simple standard test procedure was determined to evaluate the static and fatigue face-sheet peel strength. A growth model for face-sheet debonding is being checked against the test results, guidelines are proposed for improved design and a finite element model for design analysis is being developed.

Specimens were made with dimensions of 300 x 75 mm, including a Nomex honeycomb core with a thickness of 30 mm, and a cell width of 4.8 mm. Two different face sheets were used: a single ply [±45] fabric (913C-926-40), and a two ply [0/90,0] laminate with a 0/90 fabric ply (913C-926-40) and a 0 tape ply (913C-HTA-10-40). The bonding of the face sheets to the core was accomplished with an adhesive carrier layer (916C-120-55), which is a glass fiber fabric preimpregnated with a special resin to form an adhesive meniscus. Other parameters were artificially manipulated bond-line quality, and the use of a grid of adhesive film strips at the face-core interface, intended to serve as crack stoppers. Three types of experiments are being carried out: drum-peel tests, quasi-static normal peel tests, and dynamic cyclic normal peel tests (fatigue). The test set-up for the drum peel test is shown in figure 3. During the test the direction of the load remains constant. This test method is intended to determine the growth rate of debonded areas relative to cyclic normal peel loads. During the quasi-static normal peel test the specimen is loaded by an enforced displacement of the face sheet normal to the skin-to-core bonding plane. This test set-up is shown in figure 4. In this manner the bending load at the crack front is kept nearly constant. This test method is intended to determine the debonding onset loads and critical energy release rates G_k. The fatigue normal peel specimens are also loaded normal to the skin-to-core bonding-line, but by a sinusoidal force introduction with constant amplitude (see Fig. 5). Parameters are k, with $F_{dyn} = k \cdot F_{stat}$, and specimens filled with water. Forces are recorded, as well as the debonding length (with electric resistance coils), strains and displacements.

According to the first dynamic test results, the orientation of the hexagonal core material is important with respect to delamination propagation: when the major part of the meniscus orientation is perpendicular to the crack, a higher resistance is found than when this orientation is parallel to the crack. The effect of artificial aging is not as pronounced as thought, while the effect of adhesive "crack stopper" strips is very significant.

Finite element analyses of the debonding sandwich plate are being performed with ANSYS, to calculate deformations and resultant forces. Debonding is modeled by uncoupling particular degrees of freedoms at common nodes of face sheet and core adhesive layer. Results are post-processed to predict growth onset, using a failure criterion based on energy release rates. The results of one-dimensional normal peel tests will allow the evaluation of the failure criterion, using G_1 and G_k only. Using partitioning methodology allows to determine the potential direction of growth, important for multi-directional loading conditions. A debonding model, when validated with one-directional peel test results, may serve to assess failures in service. If extended to a multi-directional approach, e.g. a circular debonded area, the model may become a tool for parameter studies, and may be used to assess the critical size of face-sheet debonded areas with respect to damage growth.

5 I-Stiffened wing panels

Typical wing panels found in both civil and military aircraft are often of the skin-stiffener type, with discrete stiffeners bonded to a flat skin. The design methodology for such panels requires that the damage caused by impact, and the subsequent delamination growth under loading are characterized and understood. The major load case that is considered to affect delamination growth, thereby driving the design, is static in-plane compression. Test results obtained with coupons are not conclusive in this respect, because the impact damage configuration and delamination growth in stiffened panels are often influenced by structural features, such as the presence and spacing of the stiffeners. Comparing and relating coupon test results and stiffened panel test results is important, in order to limit the cost of testing.

DRA has undertaken a research effort to characterise the effect of impact location and stiffened panel geometry on impact damage (presented in a companion paper by DRA), and to characterise the effect of structural features on delamination growth from embedded inserts. Test results obtained with stiffened panels are also being compared to results obtained with corresponding plain plates (coupons). Ten 3-stiffener panels were made, each with a length of 300 mm. The generic configuration is shown in figure 6. The skin has a quasi-isotropic $[+45/-45/0/90]_{ns}$ lay-up, while the I-stiffeners are built up with $[+45/-45/0]_{2s}$ sublaminates. Four of these panels were used for the impact damage investigation, parameters being stringer spacings (120 and 148 mm), skin thickness (3 and 4 mm) and type of material (T800/5245 and T800/924). Six panels with embedded artificial delaminations were used for the delamination growth study, all similar and made of T800/924 material, with 120 mm stiffener spacing and 4 mm skin thickness (32 plies). The parameters considered in this part of the study are insert size (35 and 50 mm), the ply interface at which the insert is positioned (between the third and fourth layer at a 0/90 interface or between the fifth and sixth layer at a +45/-45 interface, counting from the stiffened side of the panel), and the location of the insert on the panel (in the center of a bay, below the stiffener foot, or below the stiffener center line, see Fig. 6).

The six panels used for the delamination growth studies were strain gauged and painted white for Moiré interferometry. They were tested statically in compression up to failure. Afterwards the damage was assessed with C-scan and, after dissection, with optical and electron microscopy. The panel was a non-buckled design, with a buckling strain of 0.0060 equal to the ultimate working strain. The panel with a delamination inserted beneath the stiffener center line was apparently not affected by its presence, buckled at 0.0064 and the test was stopped before failure at 0.0071, validating the design.

It was found that a midbay delamination reduced the failure strength more than a delamination under the stiffener foot, because midbay delaminations tend to reduce the buckling performance of the panel more. Delamination growth from a midbay location occurs earlier and becomes more substantial, because the presence of a stiffener suppresses mode-I displacements, thereby preventing buckling of delaminated sublaminates and delamination growth. However, regardless of the damage location, all panels failed at strains well above current design strains.

A schematic representation of the midbay damage development is shown in figure 7. First, buckling of the delaminated plies occurs (in what would be the direction of the impact), followed by bending of the main laminate in the opposite direction. Subsequently, delamination growth takes place towards the stiffeners, speeding up when approaching the stiffeners, slowing down underneath the stiffener. Finally, the delamination grows underneath the stiffener and ultimate failure takes place due to skin-stiffener separation, (in one case preceded by panel edge failure because of high local buckling deformations). It was concluded that the damage tolerance of the structure could be improved by reducing the sensitivity of the stringer foot to out-of-plane stresses. The different behavior of coupons and stiffened panels was caused by the interaction of damage growth and panel buckling that occurs in panel tests while buckling was suppressed in coupon tests.

In a similar study performed at NLR, emphasis is placed on the understanding of the influence of lay-up on damage initiation and growth. The extent of the parameter study, and the effort to "catch" the damage development under loading at different stages would have required a large number of stiffened panels. Hence, a more simple specimen configuration was defined that still contains the essential design features of the stiffened panel it represents: a "structure relevant" (SR-) specimen. This specimen has to be mounted in a special support frame, both during the impact test and during the compression-after-impact test. The supporting role of the stiffener is also taken over by a support. The main objective of this study was to determine the location within the laminate stacking sequence of the major delaminations resulting from impact. These are the delaminations that are most likely to grow in a static compression test, either in a stable or an unstable mode. Further, it was undertaken to establish the relation between compression-after-impact test results obtained with small SR-specimens and with stiffened panels, and to develop guidelines with respect to lay-up for improved design of damage tolerant panels.

A baseline I-stiffened panel was designed, with soft skin, doublers and discrete stiffeners, for an ultimate load level of 2000 N/mm at a strain of 0.0055, allowing local skin buckling at a strain of 0.0044. The material used is HTA-6376. The (soft) skin lay-up consists of 18 plies in the [0/±45/90] order: [4/12/2], the doubler area is reinforced with one 0_6 and one 0_5 "plank", and the stiffener is built up with [4/4/1] laminates, as shown in figures 8-9. The local stiffener-doubler configuration is used to study damage initiation and growth, which is represented in the derivative SR-specimens, see figure 10. SR-specimens with six different lay-ups were fabricated, one of these with the baseline lay-up, as well as two stiffened panels. The six different configurations of the SR-specimens were created by using different stacking sequences for the base skin, and by selecting different ply interfaces to insert the two planks in the skin laminate. Two 320 x 700 mm stiffened panels were made according to the baseline configuration, except that the stiffener spacing of one of the panels (253 mm) is larger than that of the baseline configuration (150 mm), while the stiffener spacing of the other panel is smaller (126.5 mm). The stiffener spacing governs the local buckling behavior of the panel, which is thought to be important with respect to the damage tolerance of the panels.

One specimen of each SR-configuration was impact damaged and the major delaminations were characterized with C-scans and fractography. Impacts were applied at 35 J, using a 1 inch diameter hemispheric tup. The impact site is the flat side of the skin, underneath the edge of the stiffener, see figure 9. In a subsequent programme three specimens of each configuration will be impacted and loaded in compression up to different load levels, in order to obtain a view of the damage development. For each configuration the failure mechanism will be established and compared with the impact damage. It is believed that the failure mechanism will consist of the growth of a few major preferred delaminations, leading to buckling of the resulting sublaminates, until panel destabilization becomes critical and ultimate failure occurs.

A comparison of the damage of two different lay-ups is shown in figure 11, for the baseline configuration and for an alternative lay-up, with the 0-degree planks located deeper inside the laminate. The upper photographs of each configuration show the lay-up, with the 90-degree plies visible as white lines. The lower photographs of each configuration show the major delaminations as white lines, at a small distance (10-20 mm) from the impact site. There is a tendency for the major delaminations to follow ply interfaces adjacent to the 90-degree plies. Apparently, the positioning of the 90-degree plies within a laminate stacking sequence influences the location of the major delaminations, and thereby the thickness of the delaminated sublaminates. Thicker sublaminates have higher buckling loads than thinner sublaminates, hence, damage growth may be delayed. It is expected that the onset of damage growth will occur at a higher load for the baseline lay-up than for the alternative lay-up, because its sublaminates are thicker, in particular the one near the flat surface, the first to buckle.

The two stiffened panels were provided with similar damage. In fact, the 3-stiffener panel was provided with four damages, two underneath the stiffener edge, and two mid-bay between stiffeners. The stiffener edge damage was similar to the damage in the SR-specimens (see Fig. 12), because the support conditions of the SR-specimens allowed similar deflections under the impactor as the deflections observed for the stiffened panels: approximately 7 mm at an impact of 35 J. After the compression tests, it was concluded that apparently the stiffener edge damage was not critical. The 2-stiffener panel failed at a strain of 0.0070 at the panel end due to fiber brooming (at four times the local buckling load). The 3-stiffener panel failed at a strain of 0.0062 (before local buckling occurred) through the mid-bag damages. The mid-bay damages at 400 mm were much smaller than the stiffener edge damage at 1800 mm , so this result confirmed Greenhalghs findings that mid-bay damages are more critical than stiffener edge (foot) damages. It was also concluded that postbuckling may not be disadvantageous with respect to residual strength, as long as it is in a long wave mode with mild gradients of displacements and skin-stiffener interface stresses.

6 Damage propagation under cyclic compression loading

The strength reduction due to impact damage such as delaminations may also be of importance in the case of compressive fatigue loading. In a study performed at the DLR - Institute for Structures and Design, the phenomena leading to the onset of delamination growth are investigated using ultrasonic and acoustic information. The type of specimen used in this study is a generic blade-stiffened panel configuration. The particular objectives of this study are to understand delamination growth onset under cyclic compression loading and to develop a test method for the in-situ observation of damage growth.

Several small 2-stiffener panels (120 x 85 mm) were cut from a larger panel. The material used for the panels is AS4/8552 unidirectional prepreg. The integral panel configuration was fabricated in one shot. The lay-ups of skin and stiffeners are quasi-isotropic $[0/\pm45/90]_s$, 3 mm thick, and the stiffener spacing is 41 mm. Barely visible impact damage (BVID) was introduced with a 10 mm diameter hemispheric impactor, at energies between 1.95 J and 5.74 J, mid-bay between the stiffeners. A 5 Hz sinusoidal compression load was applied at R = 10. Ultrasonic in-situ (USIS) inspection was used to evaluate damage progression, and acoustic emission was recorded. The relation between stiffness reduction, growth of damage area, ultrasonic images and acoustic emission was determined in a study of the phenomenological aspects of (the onset of) damage propagation.

Preliminary results indicate that compressive fatigue loading causes a stiffness reduction, even without impact damage being present. A specimen, that was impacted at a low energy level of 1.95 J, showed a significant stiffness reduction after 200,000 cycles, while the damaged area had grown less than 10%. This indicates that stiffness reduction is mainly caused by the degradation of material, rather than by damage (delamination) growth.

Damage growth is related to acoustic emission. A suddenly increased level of acoustic emission after 79000 cycles, recorded for a panel that was impacted at 4.2 J and loaded up to a strain level of 0.0038, was shown to correspond with a sudden pronounced increase of damaged area. Similarly, increased acoustic emission and a significant growth of damaged area occurred simultaneously in a specimen, impacted at 5.74 J and loaded to the same strain level, during the first 5000 cycles. The onset of damage growth in a specimen is clearly related to the impact energy level used to induce damage.

Damage development near the impact site is an intermittent process of material degradation and damage growth, corresponding to periods of low and high acoustic activity, respectively. The USIS inspection method may prove to be a technique to evaluate the degradation of the material in larger structures, and to distinguish between material degradation and damage development in a fatigue loaded specimen due to impact damage.

7 Damage tolerance of stiffened composite cylinders

Damage tolerance of composite structures is also an important issue for the design of space structures. Generic "shells of revolution", loaded in compression (and torsion) are the obvious types of structure for the study of damage tolerance at the structural level. In this contribution by DLR - Institute of Structural Mechanics, the structural concept under investigation is the skin-stiffener type, rather than the sandwich plate concept. The particular objectives of this study are to determine the buckling load reduction caused by artificial skin/stiffener detachments or by a delamination due to impact, the growth of impact and other damage under static and (R = -1) fatigue loading, to extend an in-house design optimization code with damage tolerance constraints, and to develop guidelines for the design of damage tolerant composite shell structures with respect to stacking sequences and stiffener spacing. A companion paper is presented, describing the programme in more detail.

Flat plates as well as plane and curved stiffened panels and stiffened cylinders were designed with existing optimization tools, and fabricated, some with artificial delaminations. The material used is HTA7/Vicotex-M18 with 0.125 mm ply thickness. In all, twenty four 1-stringer plane panels, eighteen 6-stringer curved panels and three 36-stringer cylinders were made, all with blade (T-) stiffeners. The 160 x 380 mm plane panels had $[\pm45_2/0/90]_s$ skins and a $[\pm45_4/0_4]_s/\pm45_4$ combination for blade and flange of the stringers, with a height of 15 mm and 30 mm flange width. The 419 x 620 mm curved panels (R = 400 mm) were made with two different skin layups, $[90/\pm45/0]_s$ and $[\pm45_2/90/0]_s$, and different thickness. The stringers (with tapered flanges) were made of $[\pm45_3/0_6]_s/\pm45_3$ (blade/flange) laminates, with a height of 14 mm and 38 mm flange width, spaced apart at 70 mm. The stiffeners were either co-cured with the skin or secondarily bonded. The R = 400 mm diameter cylinders, 780 mm high, were filament wound, with similar lay-ups and stringers as used for the curved panels. The specimens were inspected with existing ultrasonic equipment, augmented to meet the requirements for the curved panels and cylinders.

Impact tests were performed with a novel pendulum type of set-up, fully computerized and able to impact all types of specimens, including the cylinders. Residual compressive strength tests have been carried out in static and fatigue loading. Analytical work focussed in particular on buckling loads and modes. The insight gained with respect to damage tolerance will be incorporated in an optimization code for stiffened cylinders, which up to now can only consider buckling constraints.

Rather than analysing the initiation and propagation of stringer detachments/debonds, the analyses focussed on the reduction of the buckling load of a panel due to the presence of a stiffener detachment. Parameters were the size of the detachment as well as the stringer and skin geometry and laminate build-up. The analyses were carried out using standard finite elements, and modelling a single stiffener panel. In a previous study, designs were identified which tolerate large detachments with small buckling load reductions. All analyses were made for panels with $[0_2/\pm45/90_2]_s$ skins. Unfortunately, such designs often show that superior damage tolerance is negated by a lower undamaged buckling strength. It was concluded that increasing the percentage of ±45 plies in the skin laminate may improve both buckling strength and damage tolerance.

In the present study, the effect of different skin lay-ups was determined. First a $[90\ /\pm45/0_2]$ skin laminate was considered, with higher transverse bending stiffness. This change reduced the undamaged buckling load of the panel somewhat, but the buckling load reduction was significant even for small detachments (see Fig. 13). In a second skin laminate the 0-degree layers were interchanged with the 90-degree layers to $[0_2/\pm45/90_2]_s$; this led to a small increase of the undamaged buckling load and an improvement of damage tolerance with a higher detachment length. Subsequently a $[0/\pm45/90/\pm45]_s$ laminate was considered, i.e., with a higher percentage of ±45-plies. This increased the buckling load of the panel, but its damage tolerance tendency was similar as that of the previous panel. Finally, a panel was analysed with a $[\pm45_3]_s$ skin laminate, i.e., consisting of only ±45-degree plies. This laminate results in a superior undamaged buckling strength compared to the previous cases, but with a similar damage tolerance: up to a stringer detachment length of 40-50 mm the buckling load reduction of the panel is less than 2 %.

The first experimental results for a 1-stringer panel with a $[\pm45/\pm45/0/90]_s$ skin laminate, with the same fiber angle contribution as used for the third configuration that was analysed, but with a slightly different stacking sequence, are shown in figure 14. A comparison of the experimental curve and the computed curve in figure 13 for laminate 3 shows that a fairly good analytical prediction can be made for the damage tolerance of the panel in the region that is hardly affected by detachments. More analyses and experiments will be carried out to obtain a validated design procedure. However, analyses will remain limited to bifurcation buckling studies, rather than being extended to postbuckling studies including 3-dimensional effects near the detachment boundaries. It is thought that such studies are too expensive for design analysis and parameter studies. The drastic buckling load reduction computed with a bifurcation buckling analysis has clearly indicated the damage tolerance of a design, in relation to the cross-sectional dimensions and lay-ups of the panel. The following overall conclusions can already be drawn. As long as the stringer remains straight during buckling, the effect of its height on the damage tolerance is minute. Overall buckling is less affected by stringer detachments. Reducing the stringer thickness or flange width may improve the damage tolerance slightly, just as increasing the stiffener spacing, but these changes reduce the undamaged buckling strength. Increasing the number of ±45-plies increases the buckling load, especially when they are located near the skin surfaces, and can have a positive effect on the damage tolerance. However, only a few configurations have been checked experimentally so far.

8 Summary and outlook

Six different studies have been described that are aimed at improving the design methodology for composite aerospace structures, in particular with respect to damage tolerance. Two of the studies focus on a specific design problem: the prediction of residual strength in case of damage, with application to sandwich type structures. Two studies consider I-stiffened wing panels, and focus on the characterization of impact damage and damage growth related to structural features and lay-ups, and the deduction of the failure mechanisms involved. Two studies are carried out on generic structures, one to develop insight of damage growth under cyclic loading, and one to develop design guidelines for damage tolerant stiffened cylinders (which included R = -1 fatigue loading of derivative structural elements). The objectives of these studies were to develop new analysis methods, test methods or design guidelines, based on an improved understanding of the onset of delamination growth obtained through careful experimentation.

Fatigue loading is considered a relevant load case to incorporate in a design methodology for damage tolerance, as this subject was addressed by four of the studies. Common to all studies is the aspect of the scaling of results, that are obtained with small and simple specimens to limit test efforts, to larger structural elements. Improved design guidelines should certainly address this issue. At the conclusion of the work that is described here, which is expected in 1996, the question should be answered whether simple design methods, supported by empirical data, are sufficient to support the design of damage tolerant structures, or whether a more rigorous fracture mechanics type approach is essential, as followed by the COUPONS group (reported in the companion paper entitled "Damage Propagation in Composite Structural Elements - Coupon Experiments and Analyses").