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Damage Tolerance of Composite Sandwich Structures

January 2000

Final Report

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16. Abstract

The use of sandwich structures with composite facesheets in commercial aviation is increasing. Such structures provide redundant load paths and high specific bending stiffness. Because of their structural efficiency, light-weight designs result in thin facesheets that are susceptible to impact damage.

This study concentrated on the modeling aspects of damage tolerance of thin-gage composite sandwich structures. As such, it is a companion work to the FAA report "Review of Damage Tolerance for Composite Sandwich Airframe Structures," DOT/FAA/AR-99/49, August 1999, which contains a literature review and describes current technical challenges.

Current analytical attempts to assess the damage tolerance of composite sandwich structures first characterize the damage state. Based on the type and extent of damage, models are constructed to determine the stress and strain distributions. A failure criterion is then implemented with the calculated stress and strain state. Often components of the damage are neglected to simplify the model of the sandwich structure.

Two models to be developed are proposed. The first is a model to determine the extent of load transfer around a damaged area. Localized stiffness reductions will result in a shift in the neutral axis as well as load transfer in the damaged facesheet and to the undamaged facesheet. The amount of load transfer coupled with detailed models of facesheet stability and strength will provide a methodology to determine damage tolerance. The second model is of the specific damage resulting from low-speed impacts. This model will provide a methodology to predict the growth of such damage.

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EXECUTIVE SUMMARY

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In this study, several analytical models were developed to determine the load transfer in a damaged sandwich structure. Although these analysis tools have limitations that prevent their immediate application, they provide key insights as to technological challenges. These analyses explored many variables that are of concern in modeling sandwich structures with impact damage. Key factors identified include the need to model both the impact indentation and the associated delamination damage of the facesheet. It is also important to capture fully the orthotropic nature of the core material including any nonlinearities due to damage.

Two models to be developed are proposed. The first is a model to determine the extent of load transfer around a damaged area. Localized stiffness reductions will result in a shift in the neutral axis as well as load transfer in the damaged facesheet and to the undamaged facesheet. The amount of load transfer coupled with detailed models of facesheet stability and strength will provide a methodology to determine damage tolerance. The second model is of the specific damage resulting from low-speed impacts. This model will provide a methodology to predict the growth of such damage.

1. INTRODUCTION.

Sandwich structures provide an efficient method to increase bending rigidity without a significant increase in structural weight. Thin-gage facesheets (0.020″ to 0.045″) are cocured or bonded to honeycomb (aluminum or Nomex) or syntactic foam cores. These structures, based upon a minimum-gage thickness adequate to carry the in-plane loads, can carry out-of-plane loads and remain stable under compression without a significant weight penalty. The potential applications for sandwich structures are substantial. Transport aircraft components (e.g., control surfaces, engine cowlings, fairings, and fixed trailing edge wing panels), helicopter blades, optical benches for space applications, and nonferrous ship hulls are some of the current applications. In general aviation, skin/stiffener structures can be replaced with sandwich structures, where the design is based on several loading regimes including pressurization, gust, and landing loads. Sandwich construction figures prominently in future aerospace applications such as Raytheon's Premier I, Lockheed-Martin's X-33, and future tilt rotorcraft by Textron-Bell Helicopter/Boeing.

Damage tolerance of such sandwich structures has characteristics substantially different from conventional laminated structures. Besides typical damage concerns such as through penetration and delamination, additional modes including core crushing and facesheet debonding must be addressed. Often damage may not be uniformly distributed through the thickness. An impact may penetrate or damage only one facesheet while the other remains intact. Manufacturing flaws or in-service loads may also result in an unsymmetrical damage state. Cores tend to absorb and retain water often reducing mechanical properties as well as increasing the structural weight.

To fully realize the weight-saving potential of sandwich structures one must first understand the damage tolerance of such structures. This is essential in the design process to develop more efficient structures and to reduce the extent of in-service damage and frequency of repair.

1.1 TECHNICAL CHALLENGES.

Composite facesheets may fail as a result of an interaction among matrix cracks, fiber fracture, fiber kinking, and delamination. Sandwich structures may also exhibit core crushing and facesheet debonding. Damage may result from low-velocity impact such as tool drops or high-energy events such as ballistic penetration. Penetration may not be complete and only one facesheet may be damaged significantly causing a redistribution of stresses in the plane of the damaged facesheet. Damage inspections are more difficult because the core can mask the damage or otherwise impede the nondestructive inspection technique.

Current research can be divided into two major areas: damage resistance and damage tolerance. Damage resistance is concerned with the creation of damage due to a specific impact event. The variables include the material and lay-up of the facesheet, the type and thickness of the core material, and the boundary conditions of the sandwich structure. Damage tolerance is concerned with the structural response and integrity associated with a given damage state of a structure. The variables include the type, extent, and location of the damage. Often, to determine the damage tolerance of a component, flat laminates are impacted and the compressive residual strength measured. The nature of sandwich construction increases structural complexity. The presence of two load paths separated by a core that is responsible for shear load transfer combined with potentially unsymmetrical damage requires a better understanding of damage progression in order to predict residual strength.

For aircraft design, one technical challenge is to adequately predict the residual strength of a damaged composite sandwich structure. Discussions with airframe manufacturers lead one to believe there is insufficient understanding of the failure mechanisms and damage tolerance of sandwich structures. The structural response of sandwich structures is generally determined empirically based upon component and full-size test articles. This approach may not allow for adequate and timely design tradeoffs and can result in significant rework if problems materialize during full-scale testing.

The lack of understanding is partly due to the nonlinear behavior exhibited by composite sandwich structures. Out-of-plane deflections of the order of the facesheet thickness (rather than the sandwich thickness) can result in nonlinear response due to membrane effects. Structural features such as closeouts and tapered sections to accommodate fastening and unsymmetrical damage result in out-of-plane deformations and additional sources of nonlinearities. Thus, linear models cannot adequately predict the structural response let alone damage progression and residual strength. Without adequate models, the ability to predict damage onset due to overloading or growth of existing damage is problematic. Design methodology is thus based on limited coupon tests and full size test articles to either substantiate the design or indicate hot spots that need to be reworked.

Understanding the damage tolerance of composite sandwich structures will improve the structural efficiency of the aircraft. Moreover, the accurate assessment of damaged structures will reduce the frequency and cost of repair and thus improve the serviceability of the aircraft.

1.2 OBJECTIVES.

The goal of this report is to determine the current state of the art in the area of damage tolerance of composite sandwich structures. Of specific concern is the response of sandwich structures composed of thin-gage facesheets on honeycomb core. A second objective is to determine an approach to provide analytical tools to be used in the assessment of sandwich structures exposed to impact events.

2. SUMMARY OF PREVIOUS WORK.

2.1 LIE.

Lie used a combination of empirical and analytical methods to predict the residual strength of impact damaged composite sandwich panels [1]. This method treats the damaged facesheet as a beam on an elastic foundation (representing the core). The critical buckling load is determined for an undamaged panel using an Euler buckling analysis. This undamaged buckling load, P_{cr}, is correlated to an undamaged buckling stress, σ_{cr}, equation 1, where A is the cross-sectional area.

$$\sigma_{cr} = P_{cr}/A \tag{1}$$

The buckling stress for a damaged panel is determined by considering the notch sensitivity of the facesheet lay-up. For notch insensitive lay-ups, a net cross section equation is used to determine the critical damaged buckling stress (equation 2). For notch sensitive lay-ups, the Mar-Lin relation (originally developed for tensile studies) is used to determine the buckling stress (equation 3). These two equations provide an upper and lower bound for the buckling load of impact damaged sandwich panels.

$$\sigma_{cr,dam} = \sigma_{cr,undam} \frac{A_{dam}}{A_{undam}} \tag{2}$$

$$\sigma_{cr} = H_c (d)^{0.28} \tag{3}$$

where d is the damage width and H_c is the Mar-Lin coefficient.

For in-plane compression, the experimental data revealed that facesheets built with ±45° fibers are notch insensitive while facesheets built with 0°/90° fibers are notch sensitive.

The method developed by Lie is successful in providing a simple analytical method for predicting the residual compression strength of impacted sandwich panels; however, the method does not include many of the important mechanisms of the compression failure. This method considers only the size of the damage and is not capable of providing insight into damage growth.

2.2 CAIRNS.

To analyze the compression after impact performance of sandwich panels, Cairns treats the damage area as an elliptical compliant inclusion in the anisotropic facesheet [2]. The impact event is assumed to cause a reduced stiffness to the damage area. Only the effects of the damage on the facesheet are studied; the core and the back facesheet are not included in this method.

Lekhnitskii's methods for calculating the strains in an anisotropic plate with an elastic inclusion are used to determine the strains along a vector originating at the center of the damage. The Whitney-Nuismer method is used to determine an average strain value over a characteristic length (a_0) near the damage region (figure 1). This characteristic length is dependent upon the material properties and lay-up of the facesheet. The strain-ratio *(S.R.)* is calculated using the Whitney-Nuismer strain and the far field strain (equation 4).

$$S.R. = \frac{\varepsilon_x^0}{\frac{1}{a_0}\int_0^{a_0} \varepsilon_x(x,y)\,dr} \tag{4}$$

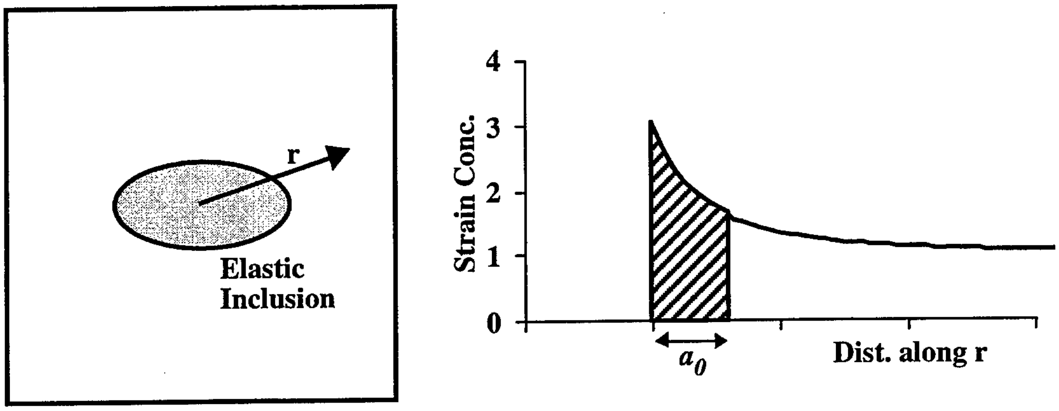


FIGURE 1. WHITNEY-NUISMER STRAIN CONCENTRATION

The vector direction resulting in the lowest strain ratio is considered the controlling case. The ultimate strength of the damaged panel is dependent upon the lowest determined strain ratio and the ultimate strength of the undamaged panel (equation 5).

$$\sigma_{ult,damaged} = S.R. \times \sigma_{ult,undamaged} \tag{5}$$

The method developed by Cairns is useful when looking at the effect of a compliant region in a damaged facesheet; however, the method used is only applicable to sandwich panels in tension.

2.3 KASSAPOGLOU.

Early work by Kassapoglou recognized that the behavior of the damaged panel is affected by both changes in material properties in the damage area and changes in the geometry of the structure in the damage area [3]. He simplified the analysis by modeling the damage area as a single "equivalent" delamination (figure 2). This equivalent delamination is located between the ply closest to the core and its neighboring ply (typically the interply area most susceptible to delamination). The size of the delamination is chosen so that the model specimen with only equivalent delamination fails at the same ultimate load as the specimen with the actual impact damage. The equivalent delamination is larger that the actual delamination caused by the impact.

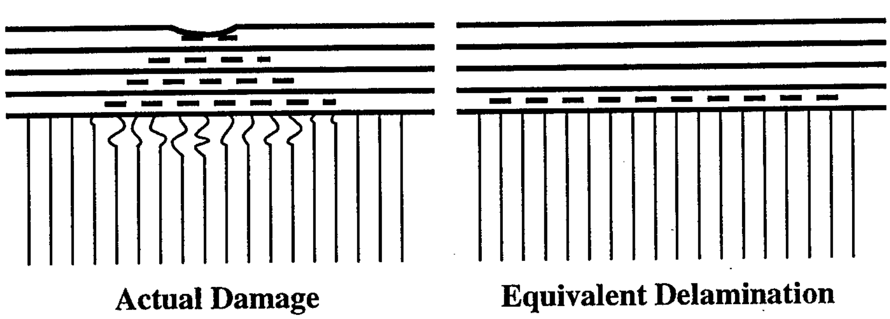


FIGURE 2. EQUIVALENT DELAMINATION

In this simplification, an eigenvalue buckling analysis [4] is used to determine the buckling load of the equivalent delamination. A postbuckling analysis is used to calculate the shear and peeling loads at all points along the edge of the delamination. A failure criteria based on the shear and peel allowables of the interply resin layer predicts failure and the onset of delamination spread. Load is increased in the panel until there is a point failure in the interply resin layer. This load is assumed to correspond to final failure.

Kassapoglou also discussed the concept of delamination threshold size, which is defined as the delamination size at which the failure mode of the panel switches from global column buckling to local delamination buckling (figure 3). This delamination threshold size is important in design as it provides the maximum delamination or disbond size that can be present in a panel without affecting the global behavior of the panel under compressive loads.

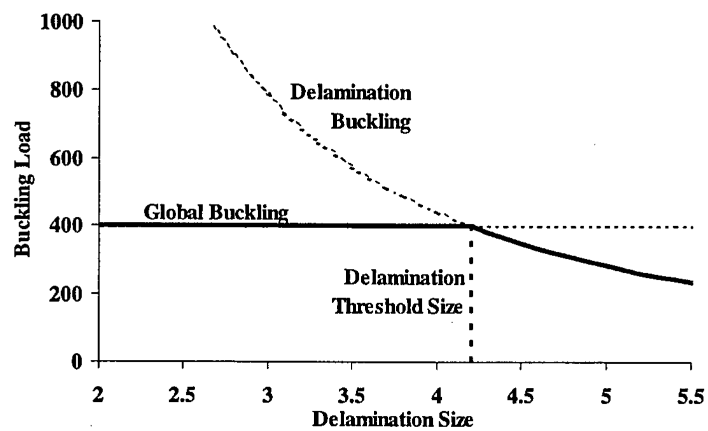


FIGURE 3. DELAMINATION THRESHOLD SIZE

Through experimentation, Kassapoglou also demonstrated that for a given material and damage size (in this case the threshold of detectability (TOD)), the ratio of equivalent delamination size to threshold delamination size is independent of lay-up [5]. To do this, the residual strength of a panel with TOD damage was determined through experimentation. These residual strength values were matched with a panel model with equivalent delamination. The delamination threshold size was also determined through analysis. The ratio of equivalent delamination size to delamination threshold size (or ET ratio) was calculated to have an average value of 1.22 (with a 7.2% coef. of variation). These values are for an AS4 fiber in a E7K8 resin with TOD damage (for various lay-ups).

The constant ET ratio for barely visibile impact damage (BVID) suggests a constant in-plane stress concentration factor (SCF), which in Kassapoglou's later work, was used to predict laminate compression failure rather than the onset of delamination growth [6]. Kassapoglou showed the SCF to be equal to the ET ratio squared (1.49 in this case). He then attempted to validate the use of a constant SCF for a specific damage level (such as BVID).

$$SCF = \frac{\text{damaged ultimate strength}}{\text{undamaged ultimate strength}} \qquad (6)$$

The SCF is calculated using methods developed by Lekhnitskii for an elastic inclusion in an orthotrophic plate. For simplicity, the inclusion is assumed to be circular. To solve for the SCF, the stiffness ratio (α) of the damaged region to the undamaged region is required. The stiffness ratio is determined by dividing the buckling stress of the largest sublaminate in the damaged area by the undamaged ultimate stress of the specimen. This is based on the assumption that the stress in the damaged region increases linearly until the largest sublaminate buckles; the stress in the damaged region then remains constant.

$$E_d = \alpha \cdot E_{ud} \tag{7}$$

$$\alpha = \frac{\text{buckling stress of largest sublaminate}}{\text{undamaged ultimate stress}} \tag{8}$$

Specimens with BVID were sectioned and examined under a microscope to characterize the properties of the largest sublaminate in the damage region. Kassapoglou's method was then used to determine the buckling stress of the sublaminate. The resulting α was used in the Lekhnitskii equations to calculate a SCF.

The results were somewhat successful with calculated SCF's of 1.21, 1.54, and 1.50 vs 1.49. The scatter in the data is attributed to several causes, including the difficulty in characterizing the largest sublaminate visually under a microscope.

In the various types of analysis procedures used by Kassapoglou, he was successful in simplifying all of the damage caused by an impact event to one parameter (equivalent delamination size or the largest sublaminate for predicting delamination growth or in-plane compression failure, respectively). These simplifications may not allow the prediction of other compressions after impact behavior, such as indentation damage growth. Kassapoglou's analysis has been successful in predicting the ultimate load for some damage scenarios found in sandwich panels.

2.4 MINGUET.

Minguet noted that low energy impact damage to thin facesheet sandwich panels results in a residual indentation in the facesheet and core buckling (crushing) beneath the impact region [7]. He also observed the compression behavior of such panels displayed a wrinkle propagation failure mode. He chose to model the damage as an initial indentation in the facesheet with a damaged core (figure 4).

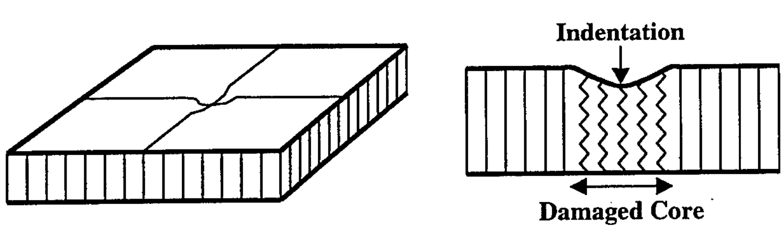


FIGURE 4. MINGUET'S DAMAGE MODEL

The facesheets are modeled using von-Karman's plate equations, and the core is included as an orthotrophic solid. The core in the damaged area is required to have zero out-of-plane stress, resulting in an unsupported facesheet. In addition, the nonlinear stress-strain behavior of the core is included in the model (figure 5).

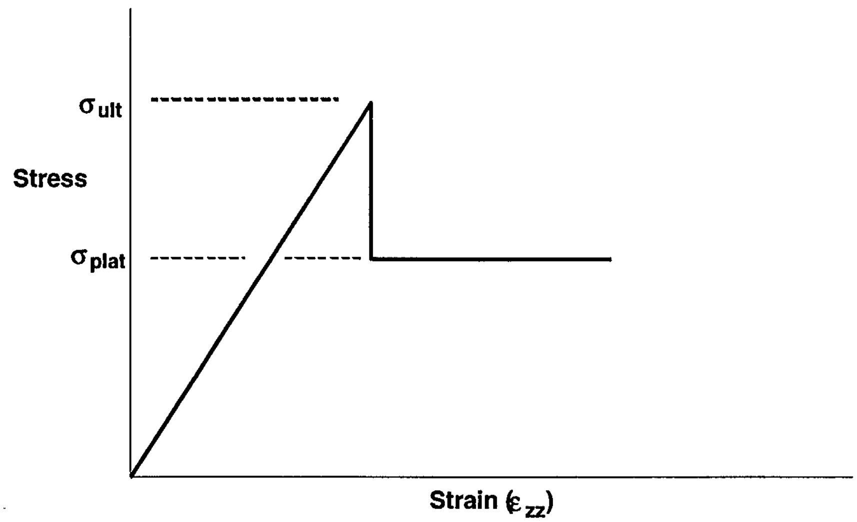


FIGURE 5. STRESS-STRAIN BEHAVIOR OF HONEYCOMB CORE

The compressive behavior of the panel is determined using a series of solutions of the displacements at the neutral axis of the front and back facesheets. Compression failure is considered to occur due to an unstable propagation of the indentation.

Minguet used some experimental data to validate the results of his analysis method. In order to obtain accurate results, the σ_{plat} values shown in figure 5 were manipulated. Minguet's model does successfully capture the lateral indentation propagation observed in the experimentation.

The modeling technique does include some important aspect of the damage properties; however, the facesheet damage (delaminations, matrix cracking) that occurs due to impact damage is not considered.

2.5 TSANG.

Tsang noted the lack of an analytical model that is useful for the prediction of the residual strength of impacted sandwich panels that considers both facesheet and core damage [8]. In Tsang's work, the influence of facesheet damage and core damage were studied.

The testing matrix consisted of undamaged panels, panels with damage caused by static indentation, panels with simulated core damage cause by static indentation (before lay-up), and panels with simulated facesheet damage caused by slits cut in the facesheets. The damaged panels and the simulated damage core were indented statically using force as a controlling parameter. Three different size indentors were used to inflict damage to the specimens.

To separate the effects of the damaged core from the damaged facesheets, the core of the panels with simulated core damage was statically indented prior to the bonding of the facesheet. Panels with static indentation were sectioned and the extent of the core damage was measured. The simulated core damage was then inflicted by crushing the core with indentors to a depth that would cause a similar amount of damage. In the damage panels, a rebound of the core was observed (caused by elastic rebound of the facesheet). This was simulated by covering the core with tape prior to the indentation. The entire panel was then cured using a special cure plate with a 3 mm hemispherical dimple that resulted in an indented, but undamaged, facesheet.

The simulated facesheet damage was inflicted by cutting two slits in the facesheet using a saw before the bond cure. The size of these slits was determined by observing the facesheet damage in the indented panels.

When loaded in compression, the damaged panels showed a propagation of the indentation perpendicular to the loading direction. The propagation is initially stable. Failure occurs when the propagation quickly spreads to the edges of the panel followed by a fracture across its width.

The panels with simulated core damage have the same failure mechanisms as the indented panels; however, the damage propagation load and failure load are higher for the panels with only simulated core damage.

A different failure mechanism occurs in the panels with only simulated facesheet damage. No out-of-plane deformation occurs before failure. Final failure is a catastophic across-the-width fracture.

A comparison of the results show that the panels with indentation damage fail at a lower stress than the panels with simulated facesheet or core damage alone. This indicates that an accurate analysis must model both the facesheet and the core damage.

Tsang used a modified version of Minguet's method of analysis to predict the residual strength of impacted sandwich panels. The core was modeled using a two-parameter foundation (rather than as an orthotrophic solid). Because the final failure had been observed to be caused by fracture, a maximum stress failure criterion was used to predict ultimate strength (rather than unstable dimple propagation). The bottom facesheet is not considered in the analysis.

Tsang's adaptation of Minguet's method is also successful at modeling the lateral indentation growth seen in experimentation. This method also has similar drawbacks in that it is incapable of including damage caused to the facesheet material during the impact event.

2.6 SUMMARY OF PREVIOUS WORK.

Previous attempts to determine the damage tolerance of composite sandwich structures begin by first assessing and characterizing the damage state. Based on the type and extent of damage, models are constructed to determine the stress and strain distributions. A failure criterion is then implemented with the calculated stress and strain state. Often some components of the damage are neglected to simplify the model of the sandwich structure.

The damage components that seem to be important are the indentation from the impact coupled with the damaged state of the impacted facesheet. The damage within these two constituents work in concert.

3. ANALYSIS.

To gain insight, several models were developed to determine the load transfer in a damaged sandwich structure. Although these initial analysis tools have limitations that prevent their immediate application, they provide key insights into the technological challenges.

3.1 GLOBAL MODELS.

A simple finite element analysis of a damaged sandwich panel was performed to gain insight into the global behavior of damaged sandwich panels under load. This simple model resulted in general information about load transfer around the damage and between the front and back facesheets.

The analysis modeled a 12- by 12-inch flat panel, as shown in figure 6. The core elements are 1 inch thick and the facesheet elements are 0.0211 inch thick.

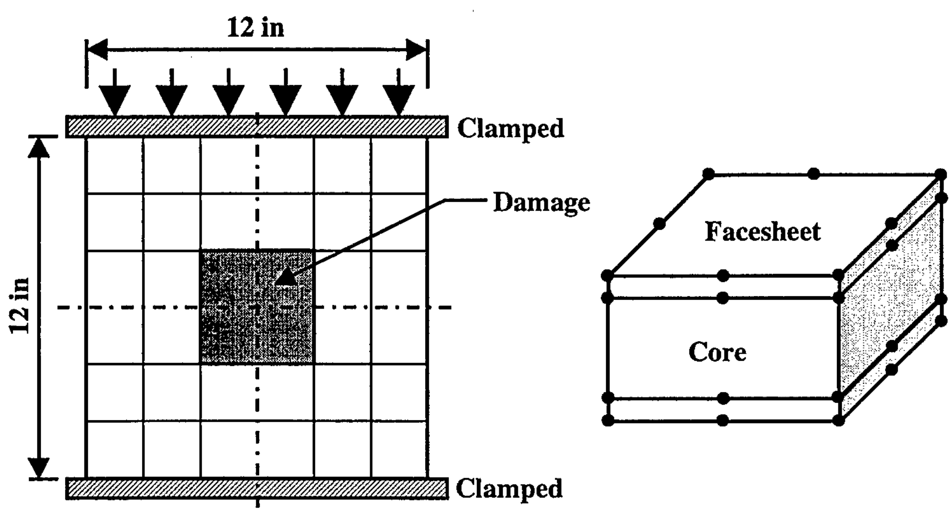


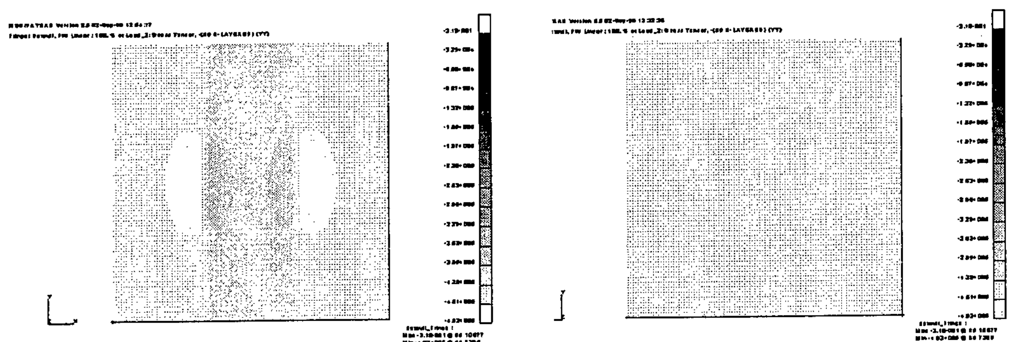
FIGURE 6. GLOBAL MODEL

Both the core and the facesheets are modeled with three-dimensional 16-node solid elements. The mesh density was varied from a grid of 3 by 3 elements to a grid of 24 by 24 elements resulting in element sizes ranging from 4 inches on a side to 0.5 inch on a side. All elements of the model were three-dimensionally orthotropic. The material properties in the model were those of the materials used in the experimental program. The boundary conditions imposed on the model were similar to those used in the testing program. The top and bottom of the panel were constrained in both displacement and rotation (clamped). The edges of the panel were not constrained (free). The load was applied as an imposed displacement of the top edge of the panel of -0.12 inch in the y direction. This corresponds to a global strain of 10,000 $\mu\varepsilon$ over the entire panel.

The impact damage was included in the model by modeling a compliant section in the front facesheet. The damage area was 16 inches square and situated at the geometric center of the front facesheet. The facesheet elements located in the damage region received a stiffness knock-down factor. This knock-down factor varied from 1% to 90% and was applied equally to all terms in the stiffness matrix of the affected elements.

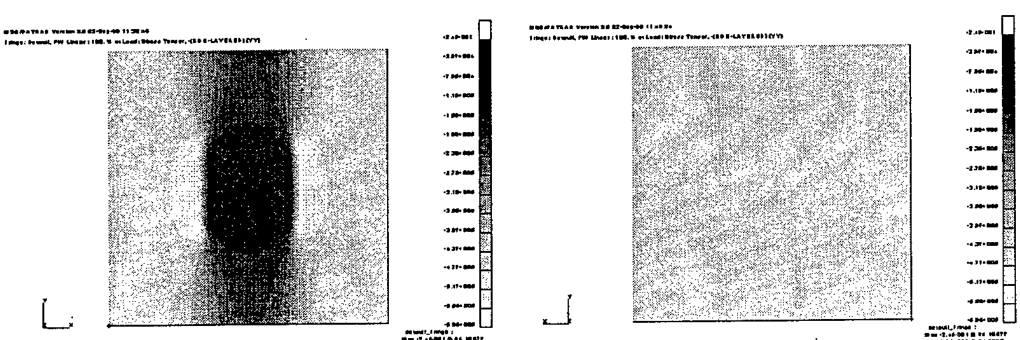
Because of the potentially large out-of-plane deflections compared to the thickness of the facesheet, a nonlinear, large deflection analysis was used to solve the model.

Analysis results showed that the total load transfer between the facesheets was small even for the large damage area used in the model. The average loads at the edge of the front and back facesheet were similar, showing little transfer between the facesheets. The global bending of the panel also appeared to be low with little out-of-plane displacement at the center of the panel. Figures 7 and 8 shows the longitudinal stresses in the front and back facesheet of a model with a damage stiffness degradation of 30% and 80%. The plot shows no stress concentrations in the back facesheet. Also, the total stress in the back facesheet did not change significantly with an increase in damage.



Front Back

FIGURE 7. LONGITUDINAL STRESS, 30% STIFFNESS DEGRADATION AT DAMAGE



Front Back

FIGURE 8. LONGITUDINAL STRESS, 80% STIFFNESS DEGRADATION AT DAMAGE

Figure 9 shows the overall effect of degrading the stiffness of the damage in the Finite Element Analysis (FEA) model as a function of mesh density. From the plot it is obvious that 3 x 3 grid is insufficient to provide reasonable results at the high percent stiffness degradation. Even with large area damage with significant stiffness degradation, there is not a great deal of load transfer at the ends of the panel.

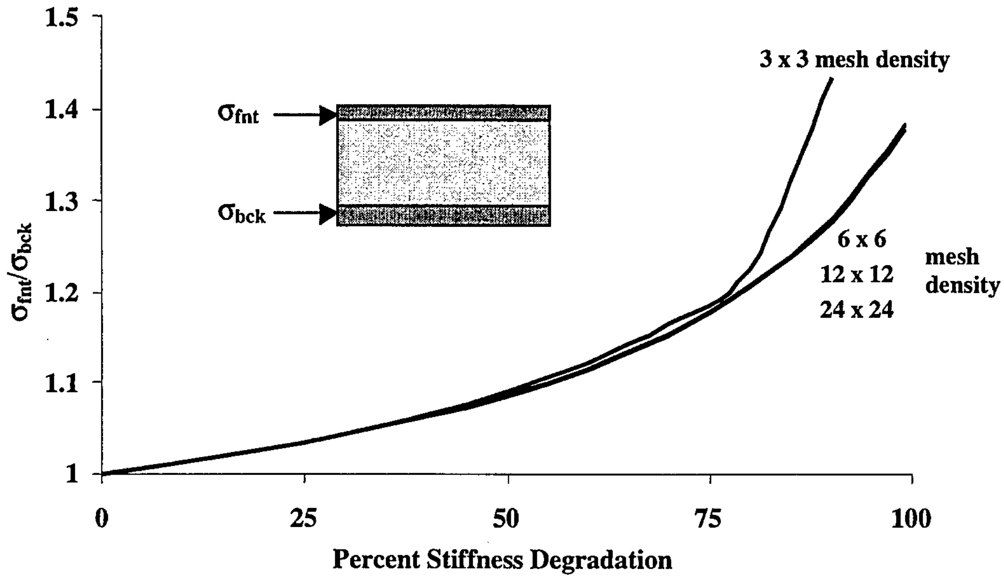


FIGURE 9. CHANGE IN LOAD DISTRIBUTION BETWEEN THE FACESHEETS WITH DEGRADING STIFFNESS

The results from the FEA agreed with experimental data [9,10]. There was no observed global bending of the test specimens under load. The far field strain gages did not show a significant difference in the load between the front and the back facesheet. These results are probably due to the high shear stiffness core transferring the load between facesheets within a short distance of the damage area. This indicates that the problem of BVID is dependent on the mechanisms of the damaged facesheet and the out-of-plane stiffness and strength of the core. Experiments show that the indentation of the facesheet and the crushed/damaged region of the core are the main contributor to the failure mechanisms, not the stress in the back facesheet.

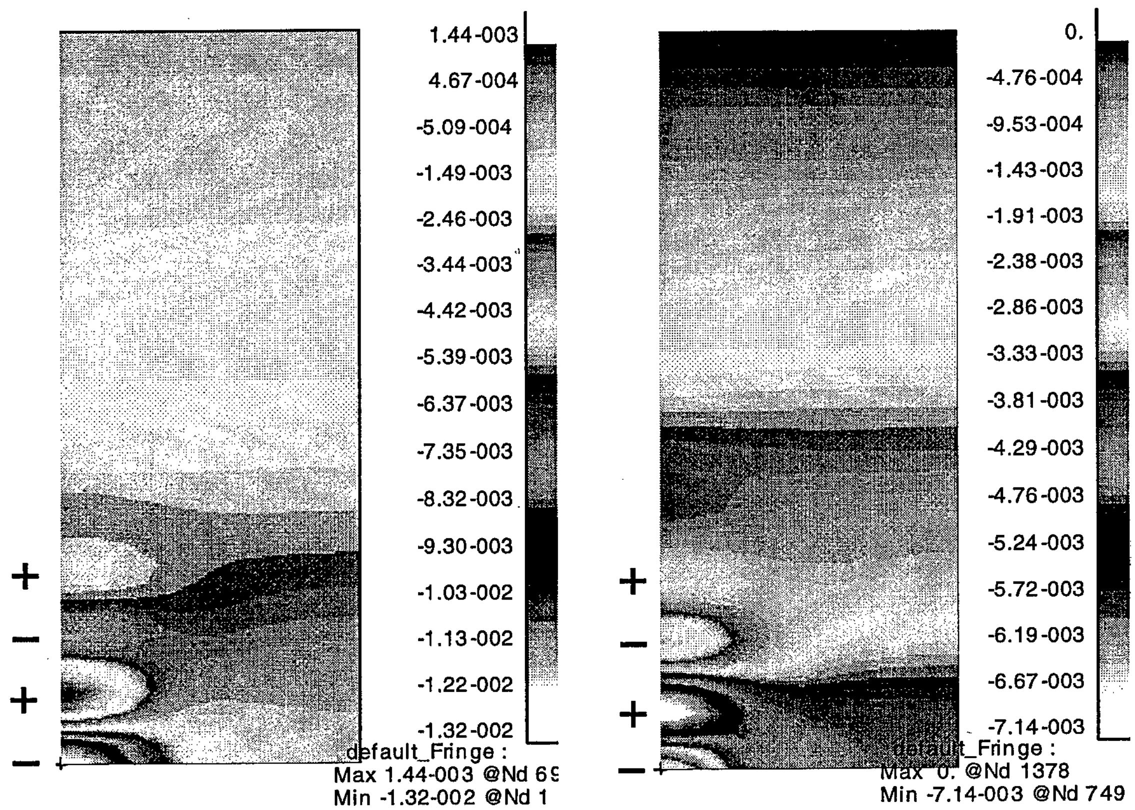
These analyses demonstrate that the failure mechanisms of an impacted panel are a local phenomenon and do not affect the global behavior of the panel or the back facesheet. In addition, in order to model the impact area, the local effects of the indentation and core damage must be included in the modelling procedure.

3.2 INDENTATION MODEL.

The first approach to modelling the compression after impact behavior of a sandwich panel using finite element methods gave mixed results. The panel was modelled using three-dimensional orthotrophic elements to represent the core and two-dimensional composite shell elements to represent the facesheets. Both the front (damaged) and back facesheets were included in the model. Clamped ends and free edges boundary conditions of the model were similar to the experimental program. Because large out-of-plane displacements were expected, a large displacement geometrically nonlinear analysis was used.

The residual indentation left after the impact event was included in the model. The facesheet nodes in the damage area of the model were located to represent the shape of the indentation corresponding to BVID. Only geometric changes were made to the model to represent the damage, there were no changes to the material properties of the facesheets or the core.

Figure 10 shows a deformed panel with the core and right half of the model hidden. The damage is on the closest facesheet. The shading represents the out-of-plane displacements of the facesheets. Lateral indentation propagation is apparent in the front facesheet. Out-of-plane displacements above and below the damage are also present. The back facesheet has out-of-plane displacements that are in phase with the front facesheet.



Front (damaged) Facesheet

Back Facesheet

+ Away From Core

– Towards Core

FIGURE 10. FRONT AND BACK FACESHEET SHOWING OUT-OF-PLANE DEFORMATION

This model does demonstrate behavior similar to that observed in the experimental program. A positive out-of-plane displacement develops above and below the initial indentation. The indentation propagates laterally with increasing load. As the load increases, the indentation propagation arrests with no further lateral displacement.

Some results from the model are inconsistent with the experimental observations. The out-of-plane displacements of the back facesheet were not observed in the tests. Figure 11 shows strain gage data from a test specimen. The gages on the back of the panel show a linear response almost to failure (gages 6, 7, and 8). The back facesheet does not seem to follow the out-of-plane displacements of the front facesheet. In addition, the onset of damage propagation occurs at a higher load in the model than observed in experimentation.

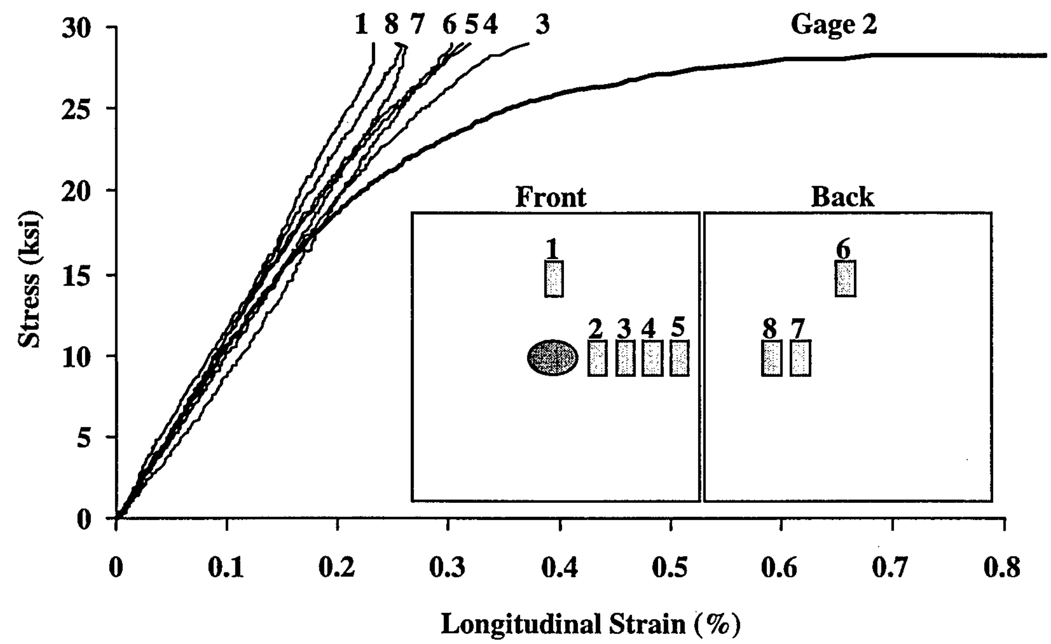


FIGURE 11. LONGITUDINAL STRAIN GAGE DATA FROM COMPRESSION TESTING

These results indicate that modelling the core as a three-dimensional orthotrophic solid may not accurately represent the local failure behavior of the core (crushing/cell wall buckling). Also, the disregard of the material property changes due to the impact event may allow for accurate modeling of the initial damage propagation.

3.3 DELAMINATION MODEL.

Often, a delamination occurs in the facesheet as a result of an impact event. The most probable place of this delamination is between the ply closest to the core and its neighbor. In order to more accurately model the damage in the front facesheet, a delamination was added to the impacted area (in addition to the residual indentation). This was modeled as a delamination between the 3rd and 4th ply of the damaged laminate. Two layers of shell elements were used in the facesheet in the damaged area to represent the two distinct layers of the delamination. The properties of the elements of the two delaminated plies represent the laminate properties of the plies they contain. The top layer included the top three plies of the laminate. The bottom layer included only the bottom ply of the laminate and was bonded to the core. The nonlinear analysis was run as before.

Figure 12 shows the model deformation. Only the front facesheet is shown. The shading represents the out-of-plane displacements and shows the out-of-plane buckling of the damaged (delaminated) ply of the front facesheet.

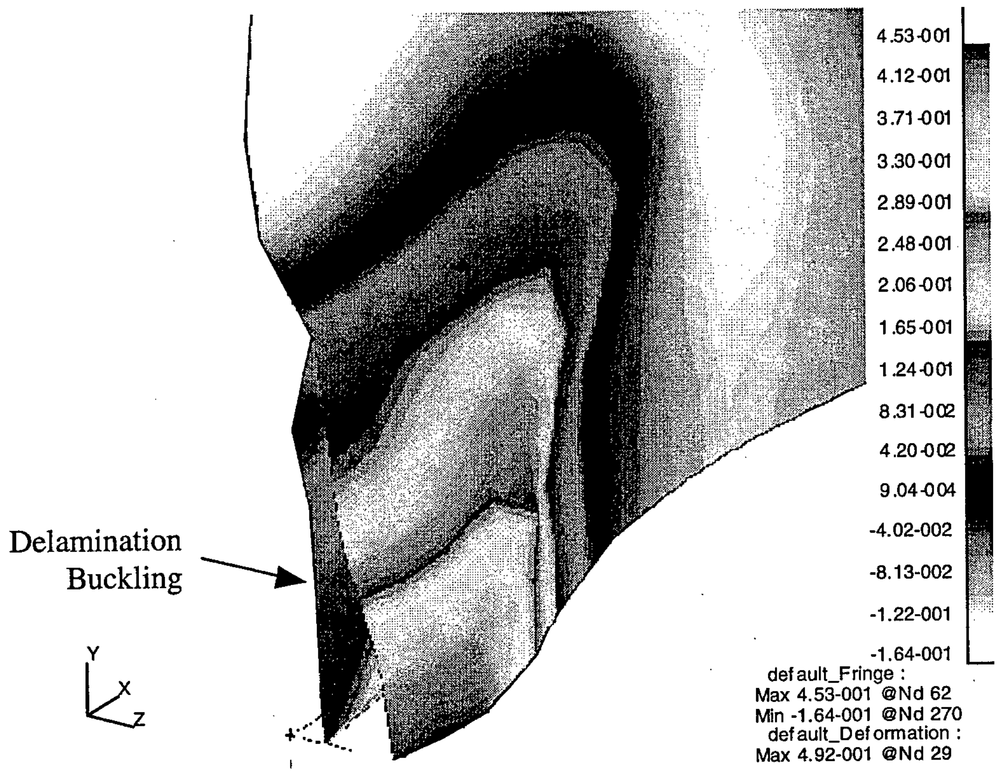


FIGURE 12. FRONT FACESHEET SHOWING BUCKLING OF DELAMINATION

These predictions do not match those observed in the experimental program. Test specimens under load demonstrated no apparent buckling of the individual plies in the damage region. It appears that the modeling of the facesheet damage must include more aspects than just the indentation and delamination to produce accurate predictions.

3.4 TSANG/MINGUET METHODS.

Miguet (Boeing) developed a method for analyzing sandwich panels with indentation damage. This method of analysis includes the most significant damage modes that contribute to the failure of the damaged panel: facesheet indentation and core damage. The damaged facesheet is modeled as an anisotrophic plate. The core is modeled as a two parameter elastic foundation. Core damage and crushing are included in the model. Stress and displacements of the back facesheet are not considered important and neglected from this analysis.

The method was successful at modeling the damage arrest mechanism that was seen in the experimental program. The energy from the compression loading changes from lateral indentation propagation to buckling along the centerline of the panel. Figure 13 shows plots of the panel displacements under progressive loading using this method.

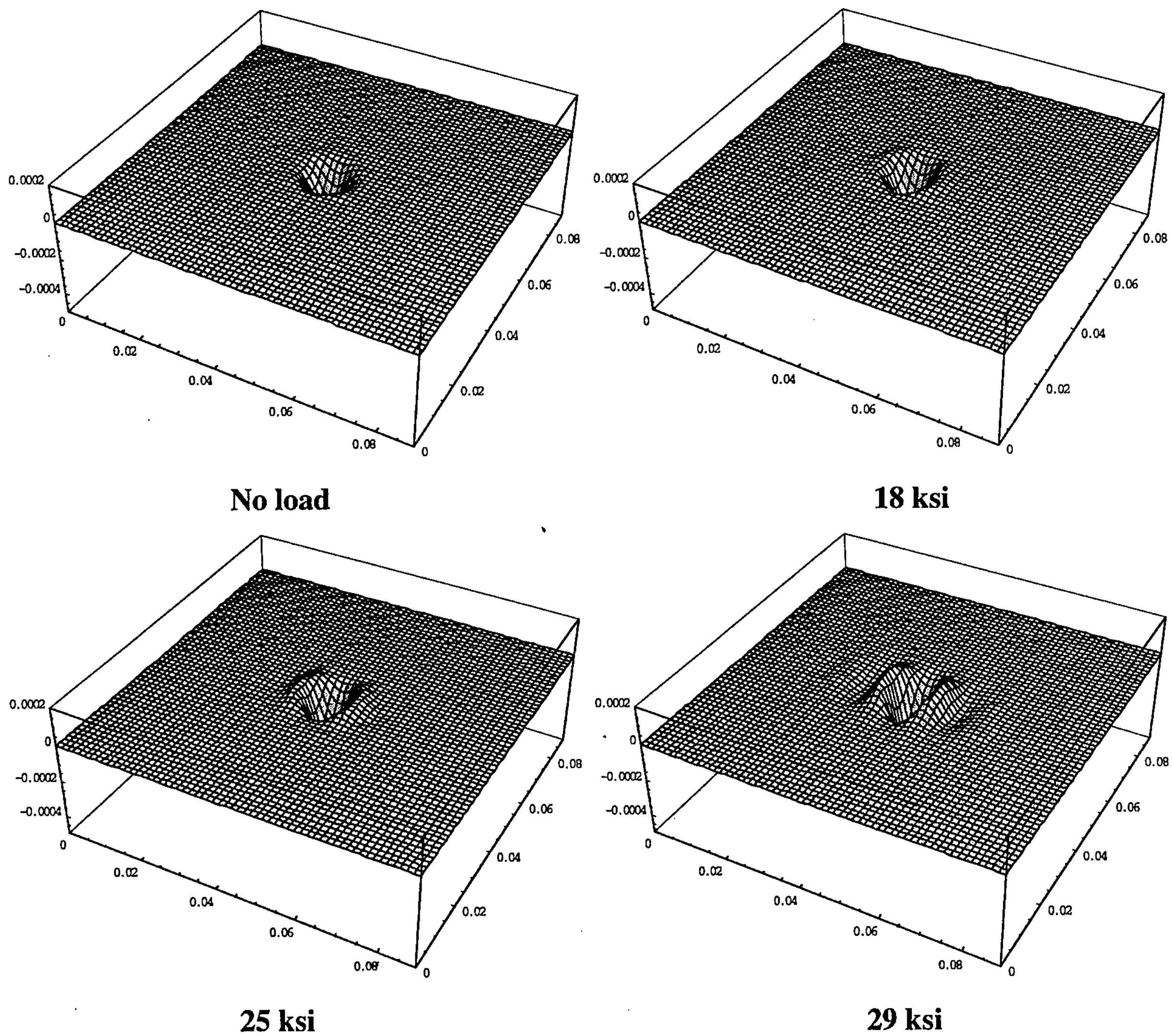


FIGURE 13. PROGRESSIVE LOADING OF INDENTED FACESHEET

Comparing these analyses with data recorded during the experimental program shows that this method does not accurately capture the onset of damage propagation. Figure 14 shows the damage propagation of both methods. This discrepancy may be caused by the influence of facesheet damage that is not included in the model. This type of indentation model can be adapted to include the effect of local facesheet damage. The damage can be modeled with a local reduction in the bending stiffness of the facesheet or by varying the thickness of the facesheet in the damage region to account for the impact damage. In addition, a delamination can be included by modeling both sublaminates of the delamination and the connecting resin layer.

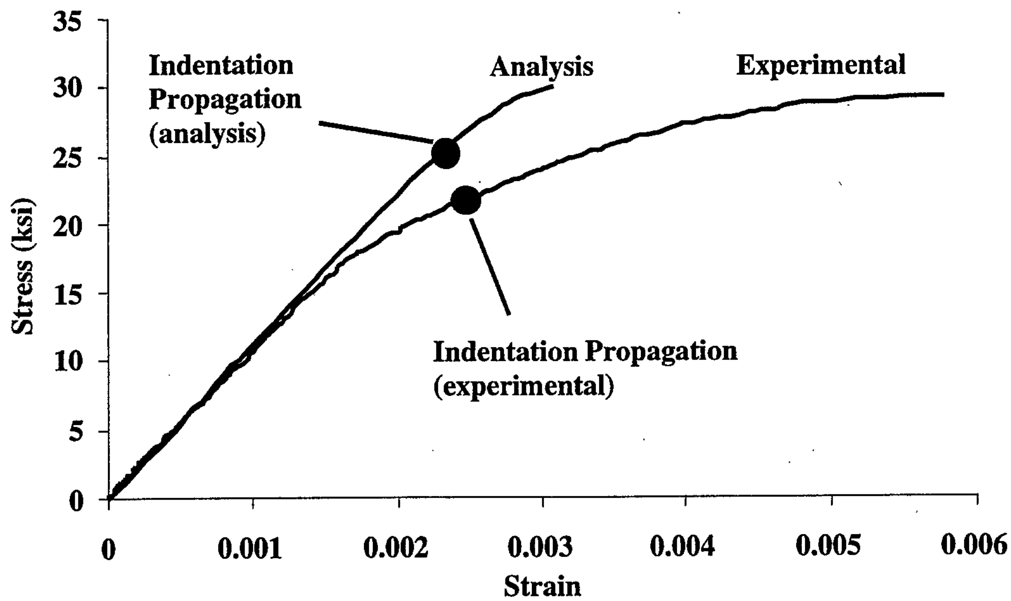


FIGURE 14. INDENTATION PROPAGATION—ANALYSIS VS. EXPERIMENT

This analysis technique is only useful for looking at a very localized area around the damage zone. The boundary conditions imposed in this method preclude its use in studying width and curvature effects.

4. CONCLUDING REMARKS AND FUTURE DIRECTIONS.

The goal of the analytical program is to establish an appropriate technique that is capable of determining the damage tolerance behavior of sandwich panels. The complex aspects of damage tolerance need to be modeled as simply as possible so that the methodology is functional; however, the method must also have the capability of accurately predicting the residual strength and life of the structure. Previous analyses have demonstrated the effects of including different damage properties in the model. Omitting the aspects of these damage properties that do not contribute to the problem and focusing on those aspects that are significant will lead to a model that is both accurate and simple to implement.

The complexities of the damage mechanisms are apparent from the experimentation. The impact damage to the core and the facesheets is very localized. Because of the localized changes in the geometry and material properties of the panel, numerical methods appear to be the best approach. Numerical methods are also flexible and can be used to study different parameters affecting the behavior.

Specifically, the research has been focusing on several things: proper modelling of the honeycomb core material, the effects of including the undamaged (back) facesheet in the analysis, simplified modelling of the material and geometric properties of the damaged region of the facesheet, and the effects of local mechanisms on global behavior.

An accurate and simple analysis method could be used to study the effects of different parameters on the ultimate strength and damage growth of sandwich panels. Important parameters may include panel width and curvature, boundary conditions, material properties, impactor geometry, and impact energy.

4.1 REDUCED STIFFNESS MODEL.

Stiffness degradation of a given damage type is to be determined. The damage types will include delamination, cracking, through penetration, core indentation, and core damage. These damage states should coincide with those that result from other studies in damage tolerance. Thus, the models to determine local stiffness reduction may be numerical and/or empirical. In addition, repaired states can also be modeled and included in the study. Repair methods also alter the local stiffness (increasing or decreasing) and can result in a strength or lifetime reduction.

In general, as different types of damage events and damage states are identified, a library of stiffness degradation tools will evolve. The methodology will thus be able to provide the amount of stiffness degradation for a wide range of damage events.

Once models are developed to determine the amount of stiffness reduction that results from a given damage state and thus a given impact event, a global model can be used to determine the load transfer about the damage and to provide a methodology to predict durability.

The numerical models will be able to assess the effect that the damage has on the load transfer. Specifically, the amount of the load that is carried through the damaged facesheet around the affected area and the amount that is transferred to the back facesheet can be determined. Local bending due to the shift in the *neutral* plane can also be determined, although this has been shown to be a negligible effect. In addition, the effect of the unsymmetric nature of the stiffness reduction can be determined as well as the need to consider nonlinear effects and the effects of boundary conditions.

4.2 LOCAL MODEL OF IMPACT AREA.

The key concern as demonstrated by Minguet and Kassapoglou is the determination of the point at which the damage due to a low-velocity impact will begin to grow. A model needs to be developed that combines the indentation aspects of Minguet's model and the reduced stiffness aspects of Kassapoglou's model. The successful combination of damage characteristics of these two models should result in a model that will accurately predict damage growth. This model will need to accurately model the orthotropic nature of the core including nonlinearity due to damage. With an appropriate database this model will provide a methodology to predict the growth of such damage.

5. REFERENCES.

1. Lie, Simon C., "Damage Resistance and Damage Tolerance of Thin Composite Facesheet Honeycomb Panels," TELAC Report 89-3, 1989.

2. Cairns, Douglas S., "Impact and Post Impact Response of Graphite/Epoxy and Kevlar/Epoxy Structures," TELAC Report 87-15, 1987.

3. Kassapoglou, C., Jonas, P. J., and Abbott, R., "Compressive Strength of Composite Sandwich Panels After Impact Damage: An Experimental and Analytical Study," *Journal of Composite Technology & Research*, Vol. 10, summer 1988, pp. 65-73.

4. Kassapoglou, C., "Buckling, Postbuckling, and Failure of Elliptical Delaminations in Laminates Under Compression," *Composite Structures*, Vol. 9, 1988, pp. 139-160.

5. Kassapoglou, C. and Abbott, R., "A Correlation Parameter for Predicting the Compressive Strength of Composite Sandwich Panels After Low Speed Impact," *Proceedings of the 29th AIAA/ASME/ASCE/AHS Structures, Structural Dynamics, and Materials Conference*, Williamsburg, VA, 1988, pp. 642-650.

6. Kassapoglou, C., "Compression Strength of Composite Sandwich Panels After Barely Visible Impact Damage," *Journal of Composites Technology and Research*, Vol. 18, No. 4, October 1996, pp. 274-284.

7. Minguet, Pierre J., "A Model for Predicting the Behavior of Impact-Damaged Minimum Gage Sandwich Panels Under Compression," *Proceedings of the AIAA/ASME/ASCE/AHS/ASC 32nd Structures, Structural Dynamics and Materials Conference*, Baltimore, Maryland, 1991, pp. 1112-1122.

8. Tsang, P. H. Wilson and Lagace, Paul A., "Failure Mechanisms of Impact-Damaged Sandwich Panels Under Uniaxial Compression," *Proceedings of the AIAA/ASME/ASCE/AHS/ASC 35th Structures, Structural Dynamics, and Materials Conference*, Hilton Head, South Carolina, 1994, pp. 745-754.

9. Moody, R. C., Harris, J. S., and Vizzini, A. J., "Width Effects on the Compression Strength of Composite Sandwich Structures After Barely Visible Impact Damage," *Proceedings of the AIAA/ASME/ASCE/AHS/ASC 40th Structures, Structural Dynamics, and Materials Conference*, St. Louis, MO, April 1999, pp. 1984-1992.

10. Moody, R. C., Harris, J. S., and Vizzini, A. J., "Curvature Effects on the Damage Tolerance of Impact Damaged Composite Sandwich Panels," *Proceedings of the 12th International Conference on Composite Materials*, Paris, France, July 1999, CD-ROM, paper #663.