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# BUCKLING, POST-BUCKLING AND FAULURE ANALYSIS OF HAT STIFFENED COMPOSITE PANEL

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## ABSTRACT

Current use of advanced composites in aircraft structures has demonstrated the potential for decreasing structural weight. But, the inherent complexities associated with the design and analysis of carbon-fiber composite structures has restricted its use in post-buckling designs. In conventional aircraft application, it is common practice to design stiffened metallic plate and shell structures to buckle below design ultimate limit. The buckling and failure characteristics of a thin composite skin reinforced by stiffeners, such as fuselage shell or wing components, have to be determined explicitly. Post-buckled design allows buckling before design limit is reached and the buckled structure can carry design ultimate load. Closed-sectioned hat stiffened panels were studied by several researchers in order to show significant post-buckling strength. Usually, the post-buckled design also results in significant weight savings when compared to buckling-resistant design. In the present study, the aim was to explore technologies for manufacturing integral composite structures to evaluate the manufacturing concepts by determining the structural behavior of hat-stiffened composite panels under compression loading. The experimental results of hat-stiffened panels for initial buckling and post-buckled response under compression loading were compared with numerical results obtained using linear and non-linear finite element analysis.

**Keywords**: buckling, compression, FEA, hat-stiffened composite panel, post-buckling

# ŞAPKA-TİPİ PEKİŞTİRİCİLER İLE TAVİYE EDİLMİŞ KOMPOZİT PANELİN BÜKÜLME, BÜKÜLME SONRASI VE ÇÖKME ANALİZİ

## ÖZET

Uçaklarda gelişmiş kompozit malzemelerin kullanımı uçak ağırlığını önemli ölçüde azaltmakdadır. Ancak, doğasından gelen karmaşık yapısı ve özellikle tasarım ve karbon fiberli kompozit yapıların analizindeki zorluklar ile birleştiğinde bükülme-sonrası kullanım imkanı önemli ölçüde kısıtlanmaktadır. Uçaklarda alışagelmiş tasarım anlayışı, pekiştirilmiş metal plaka ve kabukların en-büyük tasarım limitinin bükülme-dayanımının hemen üstünde tutulması esasına dayanmaktadır. Genellikle, bükülmeye-dayanıklı tasarımlara göre bükülmeye müsaade eden tasarım felsfesi önemli ölçüde ağırlıktan tasarrufa imkan vermektedir. Bu çalışmanın amacı, basma yükü altında çalışacak şapka tipi pekiştiriciler ile takviye edilmiş kompozit bir panelin yapısal davranışını deneysel ve FE ile analiz etmek ve bükülme sonrası kırılmaya kadar panelin ne kadar yük taşıyabileceğini ortaya koymaktır.

**Anahtar kelimeler**: basma yükü, bükülme, bükülme sonrası, şapka-pekiştICILI kompozit panel, sonlu elemanlar analizi

## 1. INTRODUCTION

The dominant part of an airframe is characterized by sheet material stabilized by stiffeners for compressing loading. The use of thin skins implies that the structure is capable of supporting load in excess of the critical local buckling load of the skin. The design of metallic aircraft structures allows certain components to buckle below the design limit load although the same confidence in the design of composite airframes has yet to be realized. Current use of advanced composites in aircraft structures has demonstrated the potential for decreasing structural weight. But, the inherent complexities associated with the design and analysis of carbon-fiber composite structures has restricted its use in post buckling designs. The buckling and failure characteristics of a thin composite skin reinforced by stiffeners, such as fuselage shell or wing components, have to be determined explicitly. Post-buckled design allows buckling before design limit is reached and the buckled structure can carry design ultimate load. Most of the researchers concentrated on the post buckling of the structures, which are reinforced with open-sectioned stiffener, such as I and J stiffeners [1-4].

Closed-sectioned hat stiffened panels were studied by several researchers [5-8] in order to show significant post buckling strength. The post-buckled design usually results in significant weight saving as well when compared to buckle-resistant design.

In the present study, the aim was to explore technologies for manufacturing integral composite structures to evaluate the manufacturing concepts by determining the structural behavior of hat-stiffened composite panels under compression loading. The experimental results of hat-stiffened panels for initial buckling and post-buckled response under compression loading were compared with the numerical results obtained using linear and non-linear finite element analysis.

## 2. FINITE ELEMENT ANALYSIS

The initial buckling, post-buckled loads and failure loads were predicted for the panels using commercially available finite element program (MARC).

### 2.1 Linear buckling analysis

The elastic buckling analysis was predicted by using an eigen value analysis based on a perturbation of the linear stiffness matrix. The eigen value analysis, which calculates the buckling loads, uses the following eigen value equation,

$$[K]\,u - \lambda\,[K_g]\,u = 0 \tag{1}$$

where K and $K_g$ are the stiffness and geometric stiffness matrices respectively, u is the nodal displacement sector, which represents the buckling mode shapes, and $\lambda$ is the load factor, when multiplying the referenced applied load, gives the buckling load as:

$$P_{cr} = \lambda P_0 \tag{2}$$

A linear eigen value buckling analysis yielded a critical buckling load of 90 kN in for the two panels. These results were in good agreement with the experimental values in the region of linear response. Both the test and analysis curves start separation at a load of about 90 KN, which indicates the initial buckling of the skin in the bay between the stiffeners.

## 2.2 Post-buckled analysis and failure

The post-buckled loads were predicted using nonlinear analysis technique. In this analysis, the length of the panel was assumed to decrease 3.3 mm at the maximum. Stepwise and each step, the following equilibrium equations were established.

$$K_t \, \Delta u^{(i)} = P_{t+\Delta t} - f_{t+\Delta t}^{\,i-1} \tag{3}$$

$$u_{t+\Delta t}^{(i)} = u_{t+\Delta t}^{(i-1)} + \Delta u \tag{4}$$

where $K_t$ is the tangential stiffness matrix at the time $t$, $\Delta u^{(i)}$ is the iterative increment of nodal displacements in the $i^{th}$ iteration, $u_{t+\Delta t}^{(i)}$ and $u_{t+\Delta t}^{(i-1)}$ are the nodal displacements at the step $t+\Delta t$ and $P_{t+\Delta t}$ is the load vector, $f_{t+\Delta t}^{i-1}$ is the vector of internal stresses at step $t+\Delta t$ and iteration (*i-1*). The Newton-Raphson Method was used to solve the above equation iteratively until a specified convergence is reached.

After initial buckling occurred in the bay it was observed that major load would be carried by the stiffeners in the way of transferring the load from the skin to the caps through the webs. When local failure occurs in any of the webs or caps, the panel as a whole was assumed to fail. In the analysis, maximum stress criterion was used to predict the failure load that means any ply stress components exceeded its corresponding strength allowable was taken as the ultimate failure load of the panel.

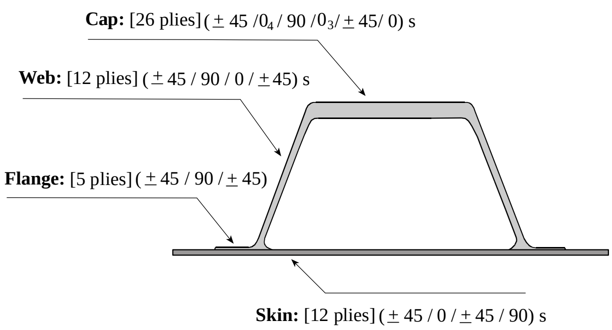



**Figure 1.** Stacking sequence of hat-stiffened composite panel

## 3. EXPERIMENTAL PROCEDURE

### 3.1 Design and manufacturing of hat-stiffened composite panel

In a previous study of stiffened composite panels [1], a literature survey revealed that the design of the ply orientation and number of plies depend on load magnitude and direction. Some of the conclusions from this literature survey are as followings:

- the outer plies should be ± 45º for all elements of the structure to carry the shear loads,
- at least one 0º ply parallel to the load should be adjacent to the outer (± 45º) plies,
- it is more efficient to place the 90º plies next to the 0º plies in order to improve both buckling and post-buckled strength.
- The use of 0º plies in the cap and skin would improve the structural efficiency, but the number of these 0º plies should be carefully calculated to meet the design loads.

- To minimize distortion, the plies should be oriented symmetrically about the mid-plane of the various elements.

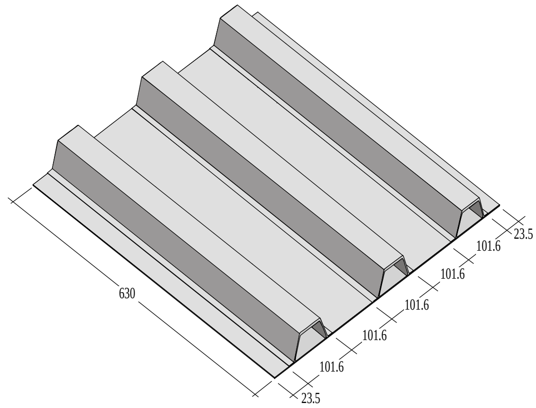



**Figure 2.** Composite panel co-cured with three integrated hat-stiffener

Based on [1] and the above guidelines, a hat-stiffened composite panel with a configuration consisting of a flat skin which is a quasi-isotropic lay-up of 12-ply and three equally spaced longitudinal hat-shape stiffeners, having a web of 12 plies, a flange of 5 plies and a cap of 26 plies, was designed, see Figure 1. Two panels, based on this design, were manufactured from Hercules unidirectional IM7 carbon fiber pre-impregnated with Cytec's Rigidite 5250-4 resin. These panels were fully co-cured with three integrated hat-stiffener, of which the design is shown in Figure 2.

Typical mechanical properties of the material used are presented in Table 1. The tooling for the panel consisted of flat surface tool plate and solid male mandrels for the hats. The solid male mandrels are made of aluminum. The tool surfaces were polished and coated with a fluorocarbon mold release to allow the mandrels to be removed from the part.

All stiffener and skin plies were cut in accordance with fiber direction configuration using templates and placed on top of each other by using lay-up technique. Due to the size of the panel and handling considerations, stiffeners and skin prepreg laminates were laid up separately and merged together in the later stage. A filler was inserted and pressed down with a roller to fit the cavity, which was formed between skin and stiffener's web-flange interface. The whole assembly was then vacuum bagged for the autoclave cure. Two hat-stiffened panels were fabricated using this tooling and co-curing technique. The panel ends were machined parallel to ensure uniform loading and potted in an epoxy resin/fiberglass mixture. The unloaded edges were not supported since the stiffeners give enough stiffness to ensure that local buckling occurred before an Euler-type global buckling mode.

**Table1:** Material Properties for IM7/5250-4 unidirectional composite

| Property | Value |
|---|---|
| Longitudinal tensile modulus | 179.5 GPa |
| Longitudinal compressive modulus | 138.4 GPa |
| Transverse tensile modulus | 10.6 GPa |
| Transverse compressive modulus | 13.3 GPa |
| In-plane shear modulus | 5.0 GPa |
| Poisson Ratio | 0.3 |
| Longitudinal tensile strength | 3.068 GPa |
| Longitudinal compressive modulus | 1.694 GPa |
| Transverse tensile modulus | 48.2 M Pa |
| Transverse compressive modulus | 257.6 MPa |
| In-plane shear modulus | 138.2 MPa |

The flat surface of the panels were painted with white paint in order to observe the Moiré fringes more easily since shadows and resultant fringes require sufficient light.

### 3.2 Testing

A compression test fixture was designed to accommodate the panels in a 50-ton DARTEC universal testing machine, see Figure 3. The two panels tested were loaded in uni-axial compression under displacement control so that catastrophic failure in compression could be assessed. Based on the literature survey [1], the maximum out of plane displacement of the stiffened composite panel would likely occur in the centre bay of the panel, near the mid-length. Thus, most of the strain gages were bonded to the mid-length area of the panel in order to capture critical strains in the buckled and post-buckled states. Back to back axial strain gages were adhesively bonded to the selected locations on the skin, cap, web and flanges as shown in figure 4.

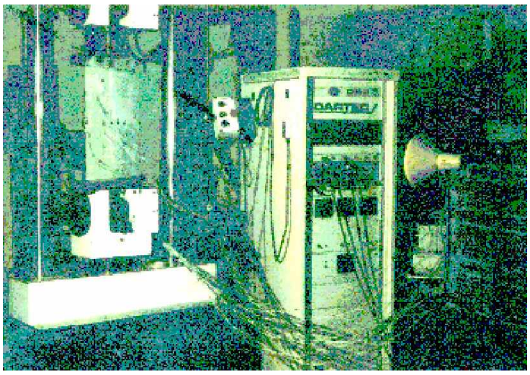



**Figure 3.** Test set-up

The compression load was increased at a constant rate until the ultimate failure of the test panel occurred. Strain and load outputs were collected and processed by a data acquisition system. The out-of-plane displacements of the buckled skin were qualitatively observed by the use of the shadow Moiré Technique.

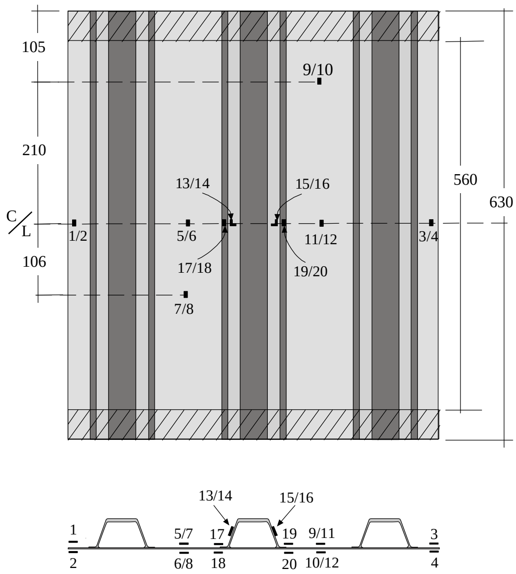



**Figure 4.** Strain gage locations on hat-stiffened panel

## 4. RESULTS

### 4.1 Experimental results

#### 4.1.1 Buckling Load

The skin bay buckled into five half-wave lengths at an applied loading of 94.3 kN. The critical load was calculated from back to back strain gages mounted at the buckle peaks. Figure 5 shows the strain results for the strain gage pair 7-8. It was noted that the mode shape underwent a gradual change characterized by an enlargement of the central half-wave length along with the formation of a waist at the centre of this buckle peak resulting from a change in local curvature. At a loading approximately 150 kN of panels, a dynamic mode jump to seven half-wave lengths was observed. see Figure 6.

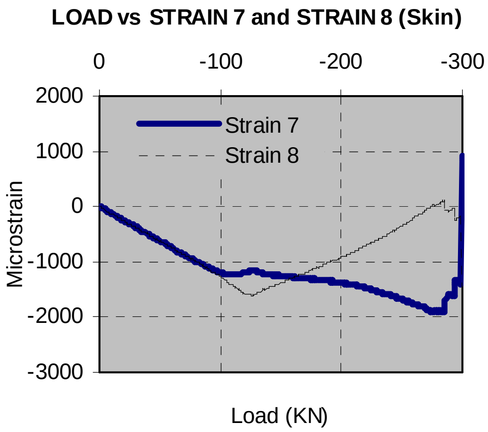



**Figure 5.** Strain in the skin (bay)

Audible acoustic emission was first detected at a loading of about 65 kN with no visible damage. This was associated with matrix cracking and more acoustic activity was detected at higher load levels. At a loading above 65 kN, visible damage, associated with transverse shear stress failure, was observed at the node-lines of the unloaded edges. This damage was localized and did not propagate in an unstable fashion with increasing load. Total panel failure was attributed to the local buckling of the supporting hat stiffeners.

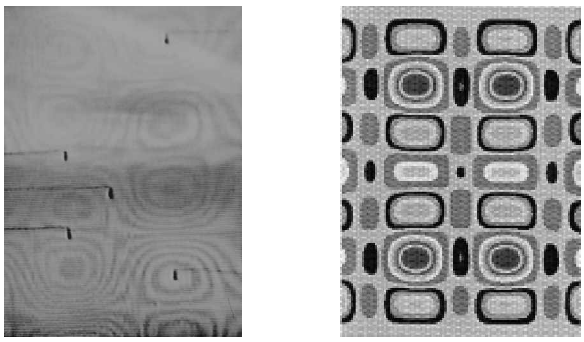
The initial buckling loads were determined from the load-strain curves as the load at which either the readings from the back to back strain gages began to separate or the point of sudden change in the slope of the curve. The initial buckling load was also estimated from the shadow Moiré Fringe Patterns and the results from these two techniques agreed well.

### 4.1.2 Post Buckling and Failure Load

The failure mode was associated with compression failure of the caps, local delamination in the flange-skin intersection and failure along the 45 º lines in the skin. The panels failed at an average load of 316 kN. The maximum strain measured in the test panel was less than 0.48 % (4804 micro-strain).

Prior to the failure of the panel, partial separation of the flange from the skin was observed as sudden change in the shadow-Moiré pattern and some audible noises were heard, but it is noted that the separation appeared to be local in nature.

The final average failure load of the panels is approximately 3.3 times higher than their average initial buckling load. This demonstrated that the panels performed with post-buckled capabilities.



(a) (b)

**Figure 6.**Mode jump from five to seven half-wave lengths (a) five buckle-peaks from shadow Moiré technique (b) seven buckle-peaks from FE analysis

### 4.2 FE Results

The skin on the two free edges buckled at a much lower load as predicted by the finite element analysis. The linear buckling analysis predicted that the local buckling loads for the edges and the bays were 66 kN and 90 kN, respectively. The average experimental buckling load in the bay was 94.3 kN.

The contour pattern in each bay was predicted as five contours at the initial buckling load and gradually evolved to seven in each bay before failure, see Figure 6. The change of contours occurred at a load above 153 kN.

Predicted failure load, based on first ply failure criterion, was calculated as 296 kN.

## 5. RESULTS AND DISCUSSION

The initial buckling load determined from the load-strain curves agreed reasonably well with the initial buckling load estimated from the Shadow Moiré fringe patterns. The predicted initial buckling load was 90 kN, which was approximately 4.5 % below the average experimental result of 94.3 kN.

The experimental average failure load of two panels tested was 316 kN and the predicted failure load was 296 KN, which was approximately 6.3 % below the average experimental result of the panels. The final average failure loads of the panels were approximately 3.3 times higher than their initial buckling loads. This verifies that the hat stiffened composite panels have a good post-buckling capability.

## 6. CONCLUSIONS

A combined experimental and finite element analysis for the initial buckling and post-buckling of hat-stiffened composite panels was presented. The predicted and experimental initial buckling and failure loads were in good agreement. First ply failure technique and maximum stress criterion to predict the failure load worked reasonably well. Panels exhibited good post-buckling strength and failed by the local buckling failure of the hat stiffeners.

## ACKNOWLEDGEMENT

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