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Testing and Analysis of Downscaled Composite Wing Box

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The fabrication, bending test, and analysis of a downscaled composite wing box were presented. Before the fabrication of the downscaled composite wing box, an I-stiffened composite plate, which was made by the same layup method as the skin of the wing box, was tested and analyzed. The graphite/epoxy wing box consisted of two cocured parts. One part had a lower skin and two I stiffeners. The other part had an upper skin, two I stiffeners, and two spars. The mold for the cocuring of the composite wing box and bending test machine of the wing box were designed and manufactured. In the bending test, the buckling mode of the upper skin was observed by a Moiré technique and compared with finite element analysis. The bending behavior of the wing box was investigated by the experiment and the finite element analysis. A good correlation between test and analysis was obtained for the bending behavior of the downscaled composite wing box.

Introduction

COMPOSITE materials have been used in secondary structures of space vehicles, missiles, and aircraft. Recently, the application of composite materials expanded to primary structures. The stiffened wing boxes made of composite materials can be applied to an aircraft fuselage or wing structures. The advantage of a composite wing box is that it satisfies the requirements of increased stiffness and reduced weight. In spite of its good performance, the use of a composite wing box is restricted, owing to the complex buckling behavior. The structural stability of a composite wing box should be verified before its wide application to the aircraft structure. In addition, it is important to investigate the buckling behavior of a composite wing box. Mechanical fastening has been used for metal aircraft structures, but cocuring or adhesive and mechanical joints are possible for composite aircraft structures. The cocuring method has recently been employed for the fabrication of composite structures because it can save cost and weight.

A limited number of papers on composite wing boxes have been reported. Skrna-Jakl et al.[1] predicted the buckling and postbuckling behavior of a composite wing box by using the finite element method. Loughlan[2] analyzed stiffened composite box sections when they were subjected to compression and bending. Romeo et al.[3] analyzed and tested composite wing boxes under pure torsion. Meyer-Piening and Anderegg[4] investigated the buckling and postbuckling behavior of a stringer-stiffened composite wing torsion box. Aston and Williams[5] presented the simplified methods for the buckling analysis of multispar wing boxes made of composite materials. Wangberg[6] tested a composite wing box under bending, torsion, and internal pressure. Greenhalgh et al.[7] described the mechanical testing and failure analysis of a multispar carbon fiber reinforced plastics wing box to improve the understanding of impact damage in a large composite structure. Ishikawa et al.[8] presented the test of a composite wing box under fatigue loading with impact damage. Suh et al.[9] tested a composite wing box prototype for a reusable launch vehicle. The literature about cocuring methods include the following. Lee et al.[10] performed a development study of a cocured advanced composite horizontal stabilizer. Kageyama et al.[11] introduced research on cocured wing structures. Scott et al.[12] manufactured a cocured carbon/epoxy aileron.

In this paper, a downscaled composite wing box and a structural test machine were made. A stiffened composite plate and a stiffened wing box were cocured in an autoclave. Nonlinear finite element analyses were performed for the postbuckling behavior of the stiffened composite plate. The bending behavior of the wing box was studied by experiments and analyses. In the experiments, the shadow Moiré technique was used to observe the skin buckling mode of the wing box. The buckling mode and bending behavior of the wing box in experiments were compared with nonlinear finite element analysis.

Fabrication

In the present study, the stiffened panel and cocured wing box are created to investigate the buckling and bending behavior of a composite structure. The geometry of the stiffened composite plate and the mold for cocuring are shown in Fig. 1. Two 0-deg plies of the flange were continuously laid up into the skin. Unidirectional tape fillers were inserted into the junction parts of stiffeners. The molds for cocuring of the stiffened composite plate consisted of three parts. The central part was made up of three submolds so that the molds were easily released after cure. This scheme was also applied to the design of the mold for the composite wing box. Graphite/epoxy prepreg was used for the stiffened composite plate and the composite wing box. Figure 2 shows the geometry of the composite wing box. It was a downscaled wing of the composite aircraft developed at the Korea Aerospace Research Institute (KARI). The chord length is reduced to a quarter, and only a small portion of full span is included in the analysis. The scaled-down wing box had two spars, four stiffeners, and two skins. Both ends of the wing box were designed for clamping. The stacking sequences of each component are shown in Fig. 3. Both stiffeners and skins had the same stacking sequences as the stiffened composite plate. The stacking sequence of the spar was $[0/90/\pm 45]_s$.

Analysis

In a previous study,[13] the postbuckling behavior of stiffened composite plates was analyzed and tested. In this paper, we analyzed the bending and buckling behavior of the composite wing box and the stiffened composite plate with the general-purpose code ABAQUS. The ply thickness was 0.125 mm in the stiffened panel. The material properties were $E_1 = 130.0$ GPa, $E_2 = E_3 = 10.0$ GPa, $G_{12} = G_{13} = 4.85$ GPa, $G_{23} = 3.62$ GPa,

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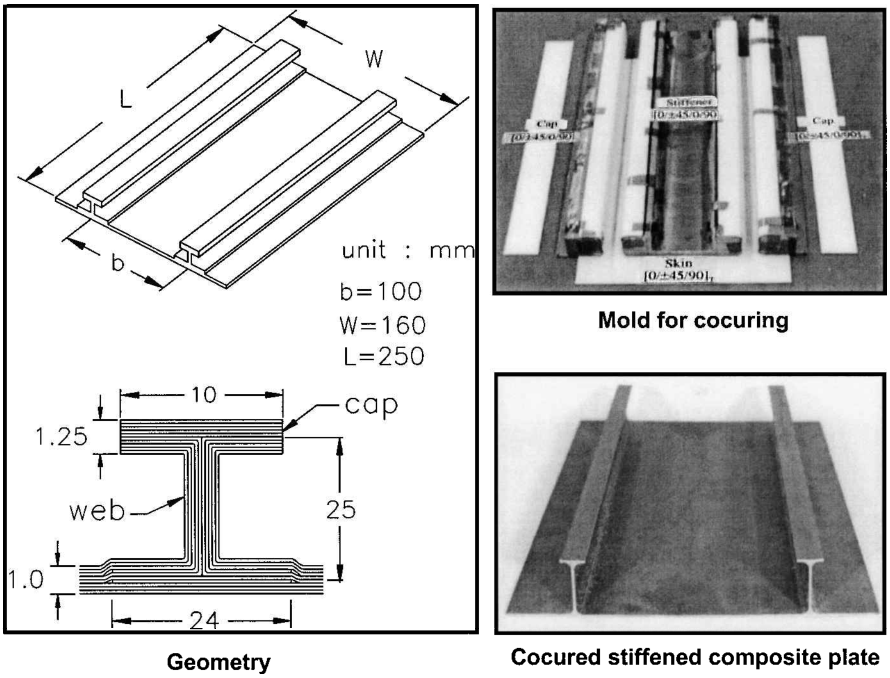


Fig. 1 Geometry and mold for cocuring of stiffened composite plate.

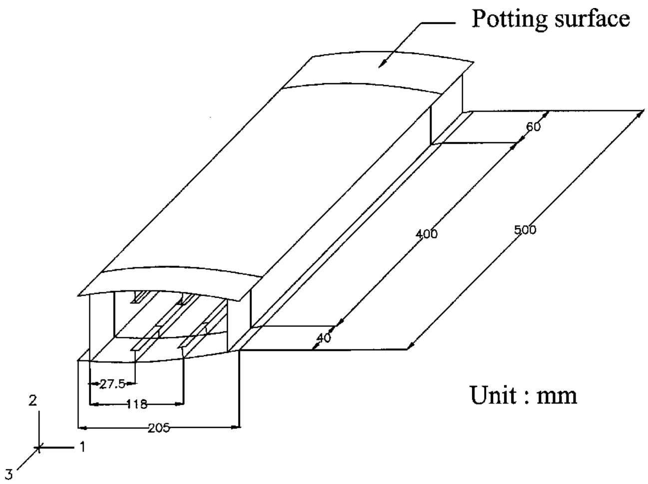


Fig. 2 Geometry of downscaled composite wing box.

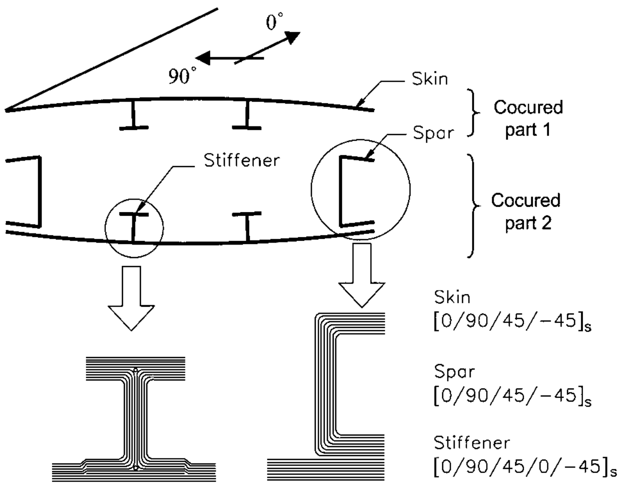


Fig. 3 Stacking sequence and cocuring parts of wing box.

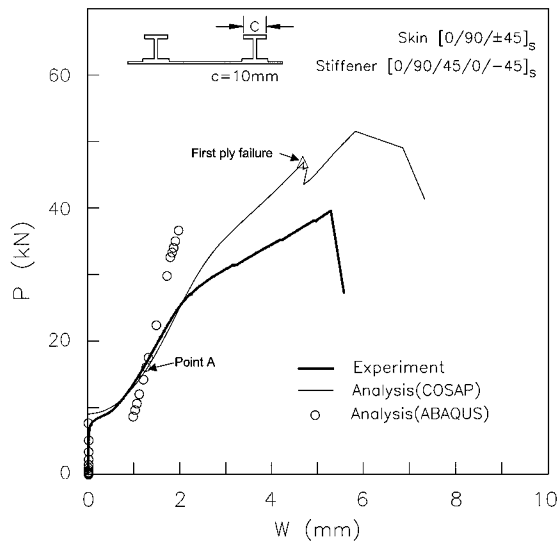


Fig. 4 Load-deflection curves of stiffened composite plate at midsurface.

$v_{12} = v_{13} = 0.31$, $v_{23} = 0.52$, $X_T = 1933$ MPa, $X_C = 1051$ MPa, $Y_T = 51$ MPa, $Y_C = 141$ MPa, and $S = 61$ MPa, where X is the strength in the fiber direction, Y is the strength in transverse direction, and subscripts T and C represent tension and compression, respectively. These material properties were obtained by coupon tests. The in-house nonlinear finite element program COSAP was used for the progressive failure analysis in the postbuckling range. The progressive failure analysis adopted the maximum stress criterion and the complete unloading failure model.[14] In the failure analysis, the considered failure modes are matrix failure, shear failure, and fiber failure. Figure 4 shows the out-of-plane deflection measured at the midskin. The postbuckling behavior and the failure analyzed by COSAP were well correlated with experimental results. The first ply failure is marked in Fig. 4. The failure mode was a matrix failure. After the first failure, COSAP analysis was close to the experimental result. We also analyzed the postbuckling behavior of the stiffened composite plate by ABAQUS. The stiffened composite plate was modeled using an eight-noded, two-dimensional shell

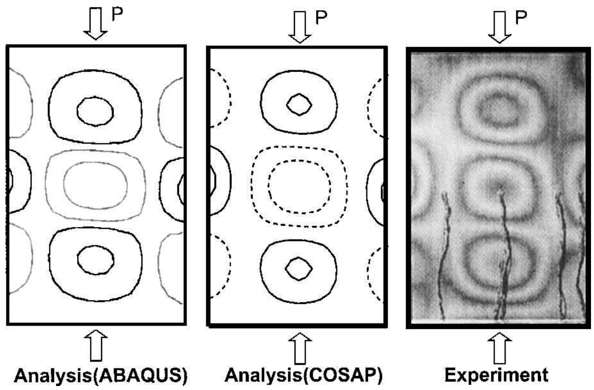


Fig. 5 Contour plots of out-of-plane deflection for stiffened composite plate.

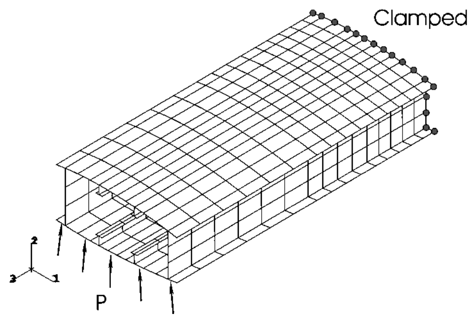


Fig. 6 Finite element model of wing box.

element, known as S8R5 element in ABAQUS. The number of elements was 288, and 2604 nodes are used in the analysis. Multipoint constraints (MPCs) in ABAQUS were applied to the loaded edges. The loaded ends were uniformly shortened using MPCs. Newton's method for nonlinear analysis was used with an automatic stiffness update method. Though the result from ABAQUS coincides well with experiment up to buckling range, there were some differences in the postbuckling region. This discrepancy appeared in the buckling region because the analysis using ABAQUS did not consider the buckling or collapse behavior, where the load-displacement response shows a negative stiffness and the structure must release strain energy to remain in equilibrium. Figure 5 shows the contour plot for the out-of-plane deflections. The buckling modes of the analyses agree well with experiments.

The investigation of the buckling behavior of the wing box was one of the main purposes of this paper. Therefore, we analyzed the bending behavior of the composite wing box by ABAQUS. Figure 6 shows the finite element model for bending analysis of the composite wing box. The buckling behaviors of the skins and the spars were predicted. The element type and the solving method were same as in the case of the stiffened composite plate. At the loaded ends, rigid elements were used to simulate potting epoxy in experiments. MPCs in ABAQUS were also applied to the loaded edges. Before we fabricated the composite wing box, we analyzed its bending behavior. The typical contour of out-of-plane displacement of the wing box is shown in Fig. 7. The load range in experiments was determined based on the analytical buckling load. A shadow Moiré technique was adopted to detect the buckling mode at the root skin.

Experiments

The configuration of the wing box mold is shown in Fig. 8. The cross-sectional airfoil shape of the wing box was that of a NACA 64_3418. The cross section of the wing consists of a multicell box beam. The mold is designed with a cocuring concept and can be used to make a composite wing box simultaneously. In this research, the two cocured parts (the upper skin and the lower skin with two spars, as shown in Fig. 3) are bonded afterward using adhesive film. Figure 9a shows the upper wing box mold, and Fig. 9b shows the lower wing box mold. The tool was made up of four basic

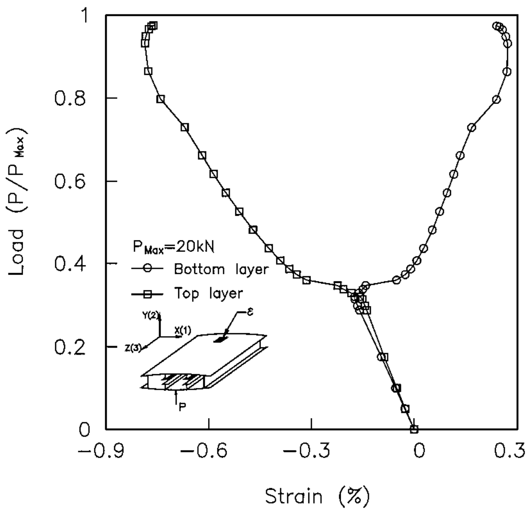


Fig. 7 Load–strain curve of upper skin.

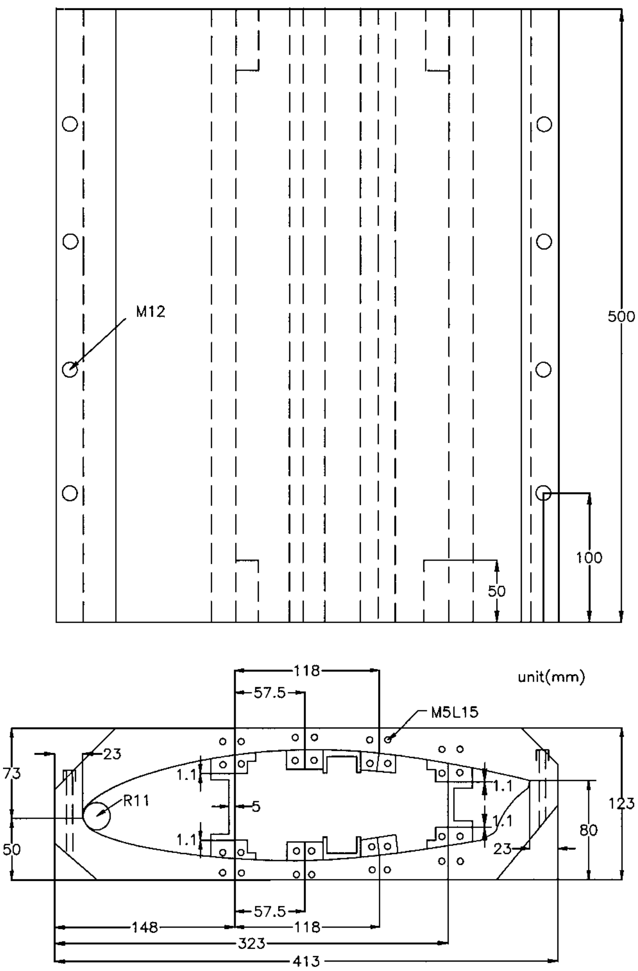
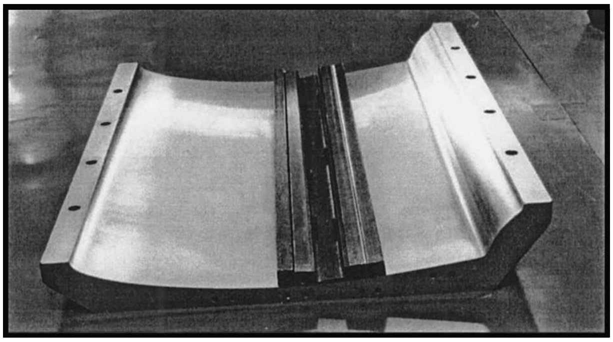
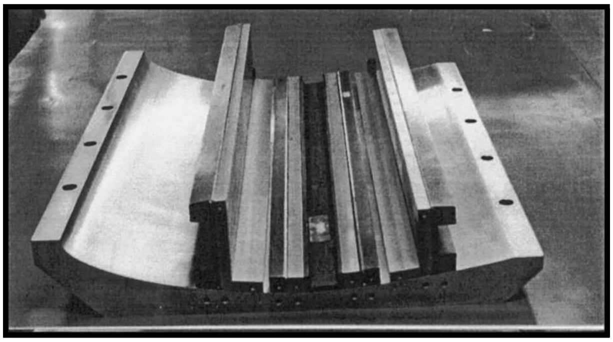


Fig. 8 Configuration of mold for cocuring of wing box.

components with a few small tools for supports and locations. The components were an upper skin mold, a lower skin mold, spar molds, and stiffener molds. The tooling material was steel, and the mold surface for the skin was ground to the shape of a NACA 64_3418. It is usually very difficult, though not impossible, to cure the whole wing box at one time in an autoclave. To cocure the whole wing box, a double envelope-type vacuum bagging technique is needed, which is expensive where labor and quality control are concerned. One possible way to fabricate a wing box using this mold is secondary bonding of all components that are cured from separate molds. The other way to fabricate is to bond secondarily two previously cocured



a) Upper wing box



b) Lower wing box

Fig. 9 Mold for cocuring.

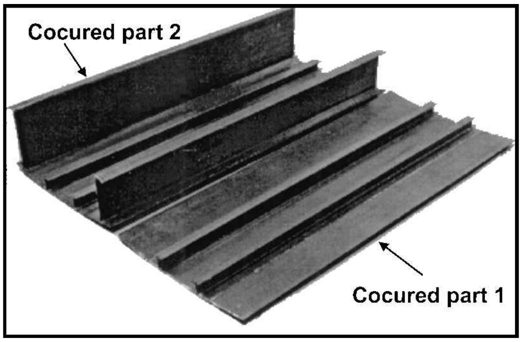


Fig. 10 Cocured parts of wing box.

parts, as shown in Fig. 3. The latter method was used in this paper. Cocuring was done for two parts. The lower cocured part integrated the upper skin, the front spar, the rear spar, and two I stiffeners. The other part was assembled by the lower skin and two I stiffeners. The composite wing box was fabricated using graphite/epoxy prepeg. I stiffeners were formed by the continuous layup of the skin to improve integrity. Release films and release fabrics were attached to each mold. After the prepeg was laid up, the assembled mold was placed inside the vacuum bag. The cure cycle was the standard cycle for graphite/epoxy composite materials. The cocured parts are shown in Fig. 10. The two cocured parts were then bonded by FM-73M adhesive film. The clamping jig was used to hold the two parts in position. The assembled parts were then cured according to the recommended curing cycle of the adhesive film. Before bonding, the surface to be bonded was polished with #150 sandpaper. The surface measured roughness for cocured parts was measured as 3 μm in the longitudinal direction and 9 μm in the transverse direction. The final shape of composite wing box is shown in Fig. 11. Two 0-deg plies were continuously laid up from the flange and the web into the skin to improve the structural integrity. Unidirectional tape fillers were inserted into the junction parts of the stiffeners. Both ends of the composite wing box were potted in the casting resin to add the role of wing rib as shown in Fig. 12. The experimental setup for the bending test of the composite wing box is shown in Fig. 13. The load cell was attached to the loading end and Moiré grid was

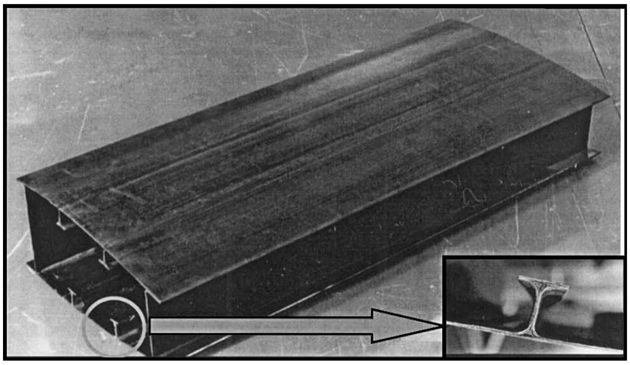


Fig. 11 Shapes of wing box and I stiffener.

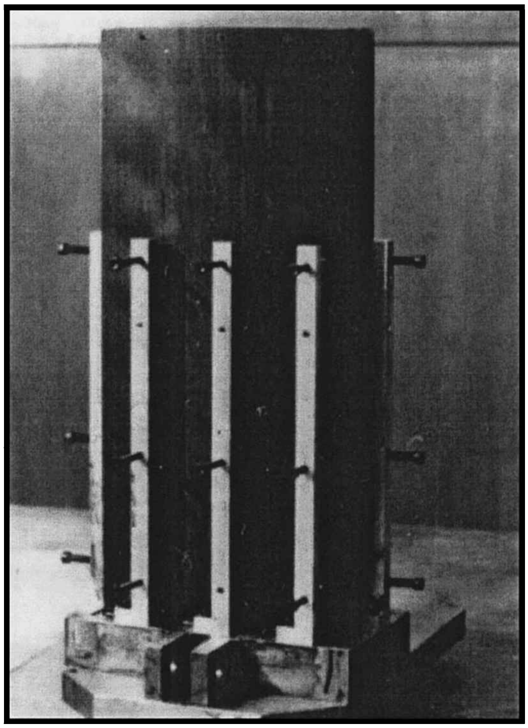


Fig. 12 Process for the potting of wing box.

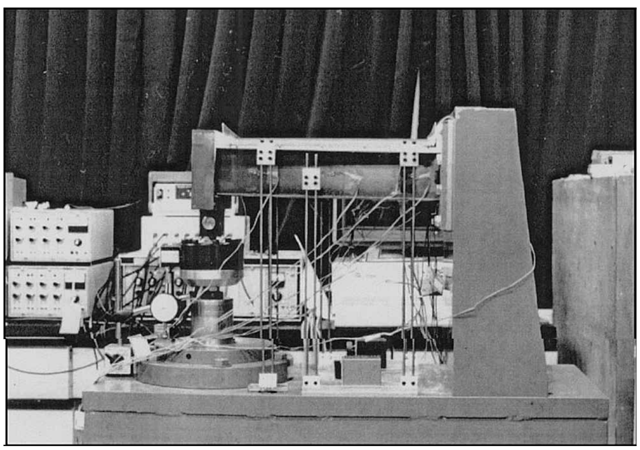


Fig. 13 Experimental setup.

installed over the root skin. The grid density was 1.7 lines/mm. A quartz–halogen lamp was selected as the light source. The velocity of the loading was 0.5 mm/min. The locations of strain gauges to monitor structural behavior are shown in Fig. 14. The locations of strain gauges to monitor buckling behavior were determined by analysis. The load–strain curve at the root skin near the rear spar is shown in Fig. 15. Finite element analysis for buckling behavior was in good agreement with experiment. However, there are some differences in Fig. 16 because of experimental imperfections such as misalignments between the test specimen and the fixture. Figure 17 shows the buckling behavior of the rear spar. The analytical buckling load of the spar was the same as that of the skin, but the experimental buckling load of the skin was lower. The out-of-plane deflection contour plots at the point A (Fig. 17) are compared for the experiment and analysis in Fig. 18. The exaggerated deformed shape is shown in the upper part of Fig. 18. The buckling of the wing box occurred between one-third of the total length and the clamped end. The experimental contour was very similar to the analysis, except at the region near the rear spar. This difference is thought to be caused by experimental imperfections near the front spar. The applied loads at the buckling instant of the stiffened composite plate and the wing box are summarized as shown in Table 1. The buckling loads of analyses agree well with those of experiments.

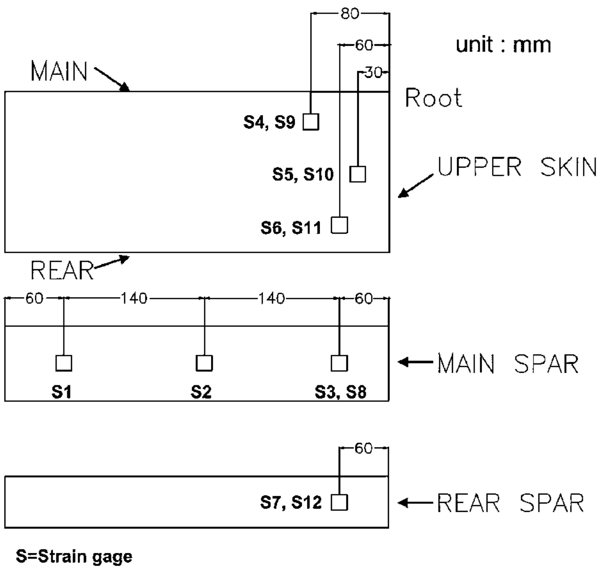


Fig. 14 Location of strain gauge.

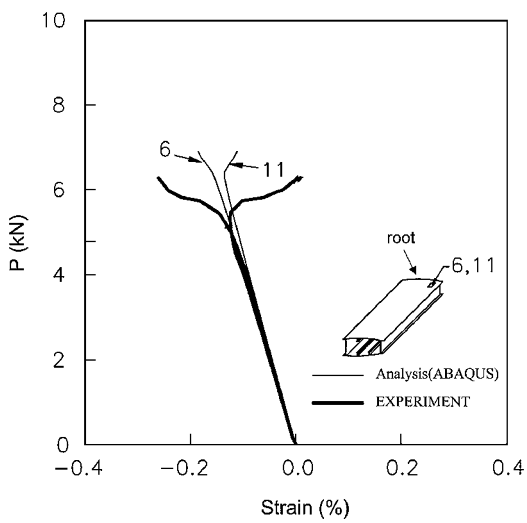


Fig. 15 Load–strain curve of upper skin.

Table 1 Applied loads at buckling of stiffened composite plate and composite wing box

Location applied	ABAQUS, kN	Experiment, kN	COSAP, kN
Stiffened plate	8.7	9.1	9.6
Rear spar of wing box	6.4	6.3	——
Skin of wing box	6.4	5.5	——

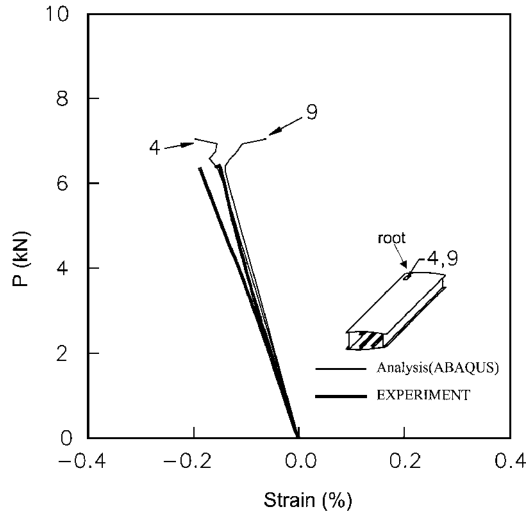


Fig. 16 Load–strain curve of upper skin.

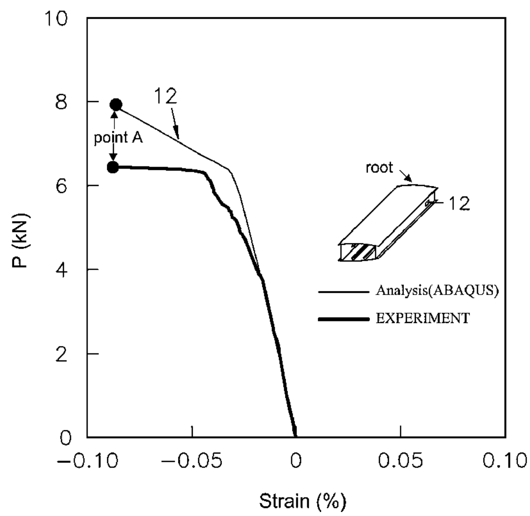


Fig. 17 Load–strain curve of rear spar.

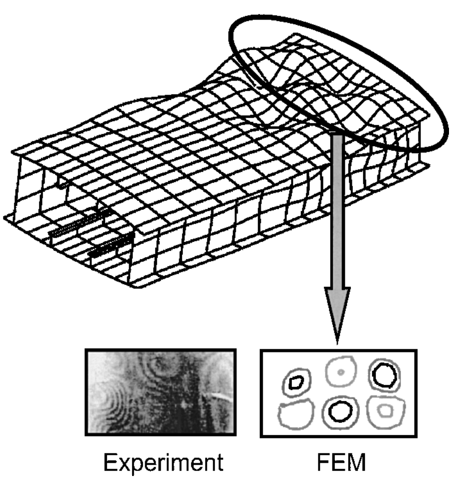


Fig. 18 Contour plot of out of deflection on the skin.

Conclusions

The downscaled composite wing box was fabricated by a cocuring method and secondary bonding. The cross section of the wing box was contoured to the shape of a selected airfoil. The bending behavior of the wing box was investigated by analysis and experiment. The postbucking behavior of an I-stiffened composite plate and the bending behavior of a wing box using a nonlinear finite element method are analyzed. I stiffeners of both the wing box and the stiffened composite plate were formed by the continuous layup of the skin to improve the structural integrity. A good agreement between the test and analysis was obtained for the wing box under bending by considering the postbuckling behavior of the skin and the spars. The buckling modes of the upper skin were observed by a Moiré technique and agreed relatively well with the results of the nonlinear finite element analysis.

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