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Composite Fuselage Shell Structures Research at NASA Langley Research Center

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Introduction

Fuselage structures for transport aircraft represent a significant percentage of both the weight and the cost of these aircraft primary structures. Composite materials offer the potential for reducing both the weight and the cost of transport fuselage structures, but only limited studies of the response and failure of composite fuselage structures have been conducted for transport aircraft. Before composite materials can be applied safely and reliably to transport fuselage structures, the behavior of these important primary structures must be understood and the structural mechanics methodology for analyzing and designing these complex stiffened shell structures must be validated in the laboratory. Methods for accurately predicting the nonlinear response and failure of structurally efficient, cost-effective stiffened composite shell structures must be developed and validated. The effects of local gradients and discontinuities on fuselage shell behavior and the effects of local damage on pressure containment must be thoroughly understood before composite fuselage structures can be used for commercial transport aircraft.

The present paper describes the research being conducted and planned at NASA Langley Research Center to help understand the critical behavior of composite fuselage structures and to validate the structural mechanics methodology being developed for stiffened composite fuselage shell structure subjected to combined internal pressure and mechanical loads. Stiffened shell and curved stiffened panel designs are currently being developed and analyzed, and these designs will be fabricated and then tested at Langley to study critical fuselage shell behavior and to validate structural analysis and design methodology. The research includes studies of the effects of combined internal pressure and mechanical loads on nonlinear stiffened panel and shell behavior, the effects of cutouts and other gradient-producing discontinuities on composite shell response, and the effects of local damage on pressure containment and residual strength. Scaling laws are being developed that relate full-scale and subscale behavior of composite fuselage shells. Failure mechanisms are being identified and advanced designs will be developed based on what is learned from early results from the Langley research activities. Results from combined load tests will be used to validate analytical models of critical nonlinear response mechanisms as well as shell scaling laws.

COMPOSITE FUSELAGE SHELL STRUCTURES RESEARCH

The objectives of the Langley composite fuselage shell structures research program are to develop the structural mechanics methodologies needed to predict reliably the response and failure of composite fuselage shell structures that are subjected to combined internal pressure and mechanical loads, and to understand the effects of local damage on the damage tolerance and residual strength of these structures. These structural mechanics methodologies include structural analysis methods, structural sizing procedures and structural scaling methods. The structural analysis methods will be used to predict the nonlinear response of internally pressurized fuselage shells and the local stress and deformation gradients that cause failure in composite shells with discontinuities. The structural sizing procedures will be used to conduct minimum weight design studies for candidate shell design concepts and to determine the sensitivity of the response and structural weight of a design to changes in structural parameters. The structural scaling methods will be used to study subscale models of candidate design concepts in an attempt to reduce the cost of design development by minimizing the amount of full-scale development testing needed for a new structural design The structural mechanics methodologies developed by this research effort will be verified in the laboratory by conducting experiments with curved stiffened composite panels and pathfinder pressurized composite shells. These experiments will also identify critical failure modes and the effects of local damage and stress and deformation gradients on composite shell behavior.

Objectives: **Develop verified structural mechanics methodologies for reliably predicting the response and failure of composite fuselage structure subjected to combined internal pressure and mechanical loads and to local damage**

Approach:

- **Develop and apply structural analysis methods that predict the nonlinear response and failure of composite fuselage shell structures with combined loads**
- **Develop structural sizing procedures and conduct parametric studies for structurally efficient composite fuselage shell structures with combined loads**
- **Develop scaling methodology for composite fuselage shells with combined loads**
- **Test benchmark curved panels and pathfinder stiffened shells to identify critical failure modes, to verify structural analysis methods, and to understand the effects of local damage and gradients on composite shell behavior**

Figure 1

PRESSURIZED COMPOSITE FUSELAGE SHELL

An important effect of internal pressure on a stiffened composite shell structure is suggested in figure 2. The relatively thin skin of a pressurized frame-stiffened fuselage shell expands outward in the radial direction more than the stiffer frames and a local bending gradient is generated in the skin where the skin is attached to a frame. The radial deflections in the thin skin can be large enough that the behavior of the shell is nonlinear. The local bending gradients will cause local three-dimensional interlaminar stress gradients in the skin which could be large enough to cause failure to occur. Inplane compression and shear stress resultants in the skin that are caused by mechanical loads may increase the magnitudes of these local interlaminar stress gradients due to the coupling of the inplane stress resultants and the out-of-plane deflection gradients associated with nonlinear thin shell behavior.

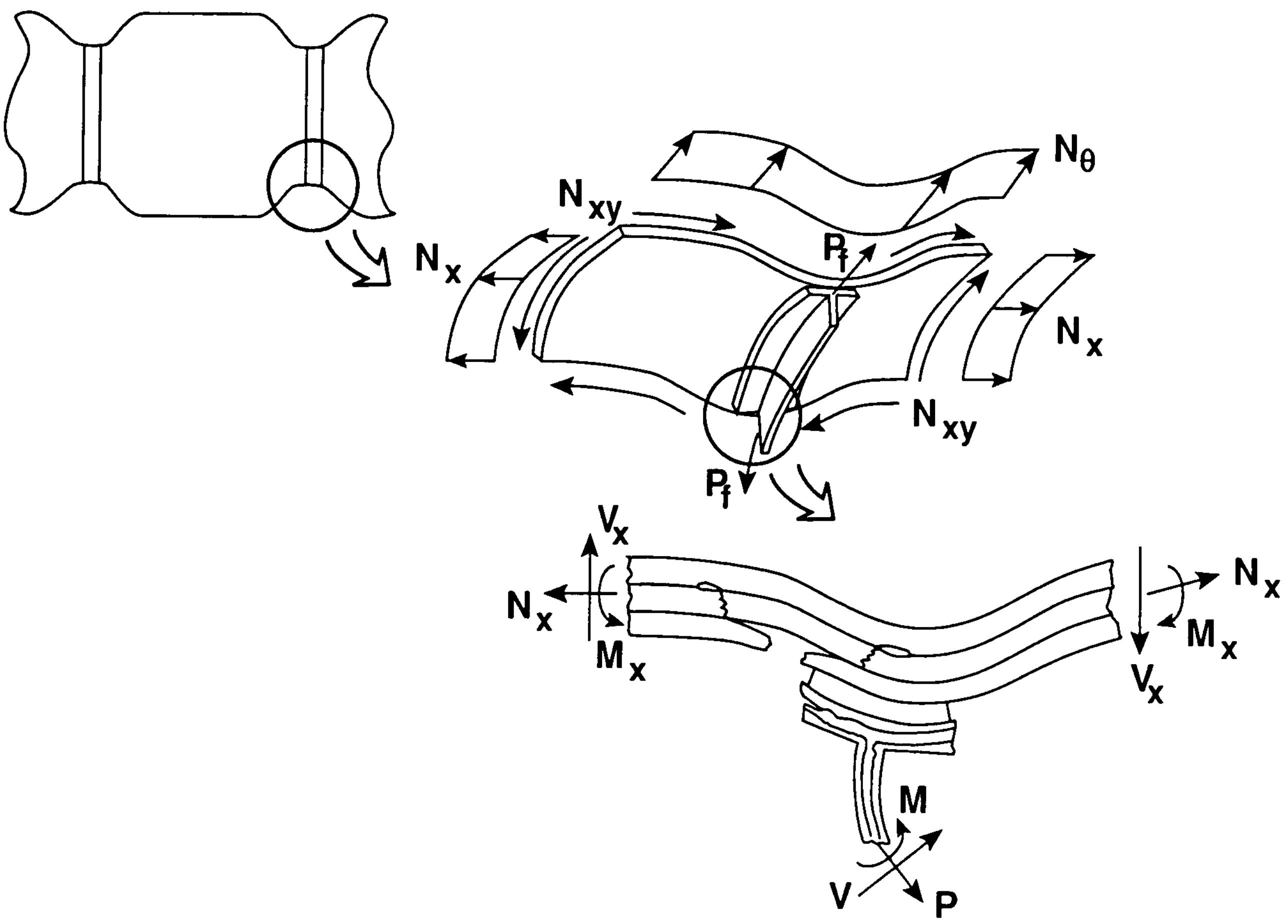


Figure 2

COMPOSITE FUSELAGE SHELL STRUCTURES RESEARCH
SHELL ANALYSIS AND SIZING STUDIES

Nonlinear shell analysis and structural sizing studies for the Langley composite fuselage shell structures research program are indicated in figure 3. The effects of combined internal pressure and mechanical loads on nonlinear structural response will be studied analytically. The postbuckling response of the skin and the redistribution of internal loads associated with stiffness changes due to nonlinear skin buckling response and damage propagation will be included in the nonlinear analyses. The local stress and deformation gradients associated with local details, discontinuities and eccentricities will be determined for accurate failure analyses and the effects of shell curvature on nonlinear behavior and local gradients will also be studied. Structural design studies will be conducted to determine minimum-weight designs for candidate design concepts subjected to combined internal pressure and mechanical loads. Studies will also be conducted to determine the sensitivity of the response and failure of candidate minimum-weight design concepts to changes in structural parameters.

- Nonlinear Shell Analysis
  - **Stiffened shell response to pressure and mechanical loads**
  - **Postbuckling response**
  - **Local deformation and stress gradients caused by local details, discontinuities and eccentricities**
  - **Curvature effects**
  - **Local stress fields for failure predictions**
  - **Internal load redistribution associated with stiffness changes due to nonlinear response and damage**
- Structural Sizing Studies
  - **Minimum-weight design studies for pressure and mechanical loads**
  - **Parametric studies**

Figure 3

COMPOSITE FUSELAGE SHELL STRUCTURES

Hierarchical shell models that represent three levels of structural modeling refinement for the Langley composite fuselage shell structures studies are shown in figure 4. Relatively coarse stiffened shell models will be used to determine the global structural response and internal load distributions due to combined internal pressure and mechanical loads. These models will also be used to determine the redistribution of internal loads due to a local stiffness change caused by skin buckling and damage propagation. Refined curved stiffened panel models will be used to determine more accurately the local gradients caused by the interaction of skin and frame elements and to predict the behavior of stiffened panel specimens of selected concepts. More highly refined shell element and structural detail models will be used to predict the local stress and deformation gradients associated with local discontinuities, eccentricities and other details and these gradients will be used to predict local failure.

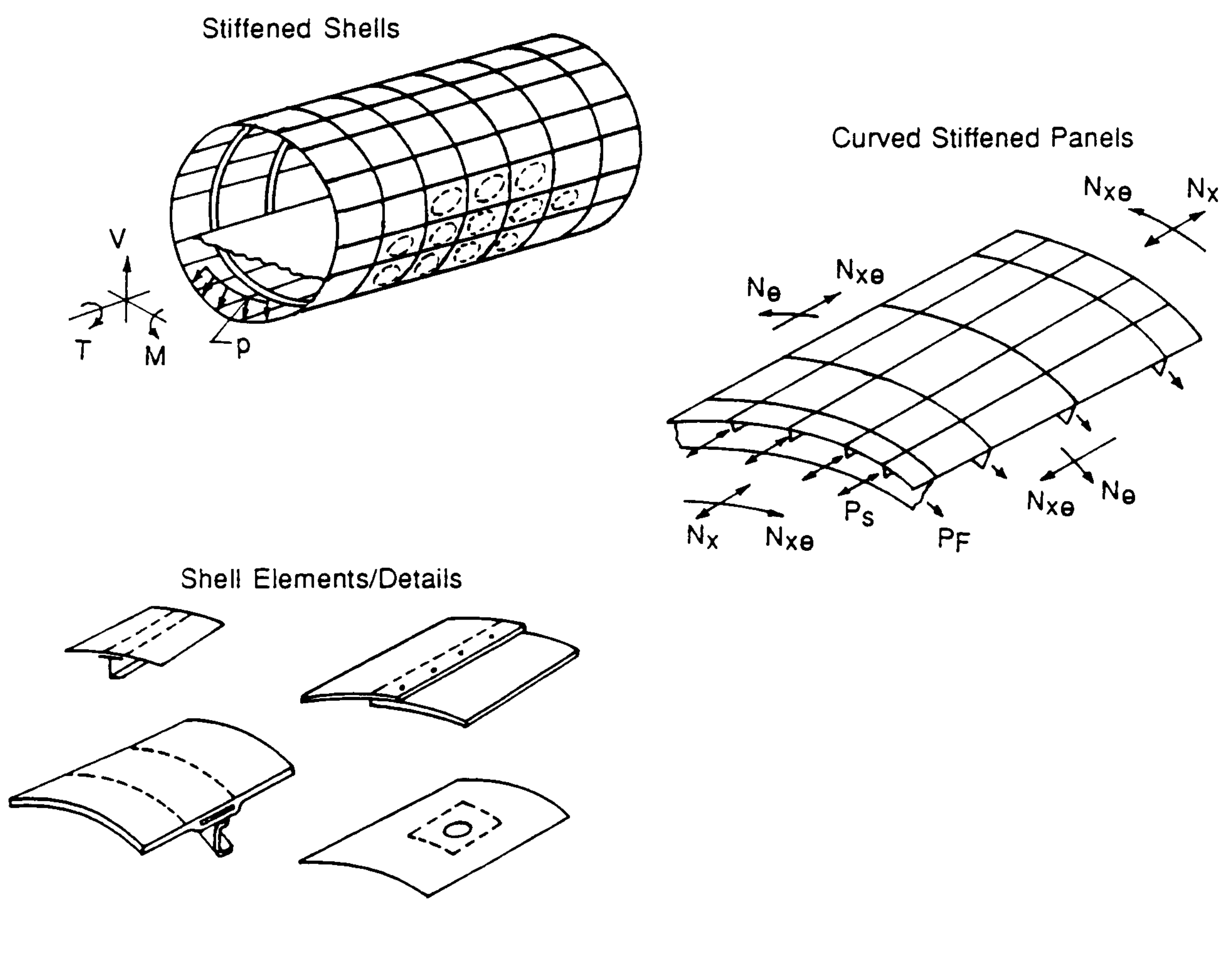


Figure 4

EFFECT OF INTERNAL PRESSURE ON COMPOSITE SHELL STRUCTURES

An example of results for a stiffened shell analysis model currently being studied is shown in figure 5. The shell model is based on the current Boeing design being developed under NASA contract NAS1-18889 and is being used to develop the Langley pathfinder half-scale stiffened shell design. The shell radius is 122 inches, the shell length is 264 inches and the shell skin is made from a [±45/90/0/±60/90]s graphite-epoxy laminate. The shell is loaded by an internal pressure of 10.35 psi. The model includes 3 skin bays with discrete stringers, frames and floor beams. The figure shows the effects of the stringers, frames and floor beams on the hoop stress resultant distribution in the skin. These results indicate that the value of the hoop stress resultant is significantly affected by the interaction of the skin and the frames, stringers and floor beams.

Radius = 122 in., Length = 264 in.
Shell Laminate [±45/90/0/±60/$\overline{90}$]s
Pressure = 10.35 psi

Reference Design Based on Current Boeing Design

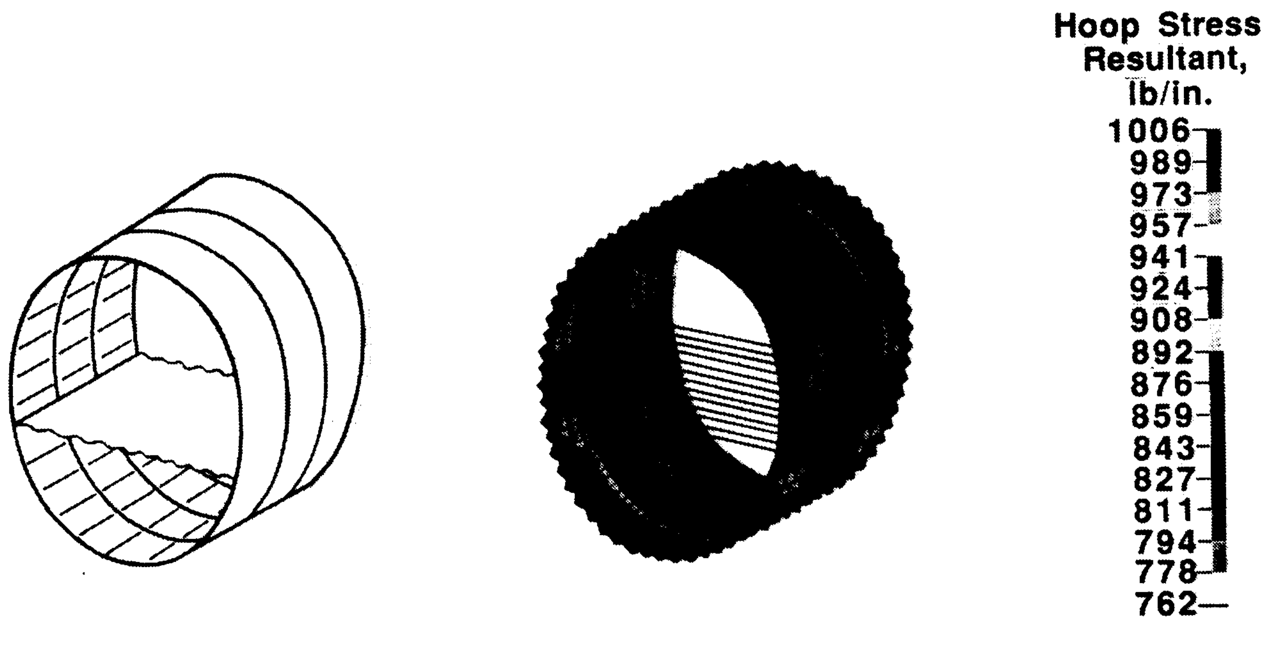


Figure 5

EFFECT OF INTERNAL PRESSURE ON UNSTIFFENED CURVED GRAPHITE-EPOXY PANEL RESPONSE

An example of the effects of internal pressure on the response of curved unstiffened graphite-epoxy panels is taken from ref. 1 and shown in figure 6. The panels are 20 inches long, 8 inches wide and have a 60 inch radius. Analytical and experimental out-of-plane deflections w at the center of the panel are shown in the lower left figure as pressure increases for 5-, 8- and 16-ply-thick panels. These results show that the panels stiffen as the pressure is increased and that the pressure-deflection response curves are nonlinear. The circumferential or hoop strain distribution along the x or circumferential coordinate from the center of the panel to a panel edge is shown in the lower right figure for an 8-ply-thick panel with 50 psi internal pressure. Inside and outside surface strain results indicate that a significant bending strain gradient exists near the panel edge. This bending strain gradient is severe enough to cause the panel to fail along this edge as shown in the upper right photograph.

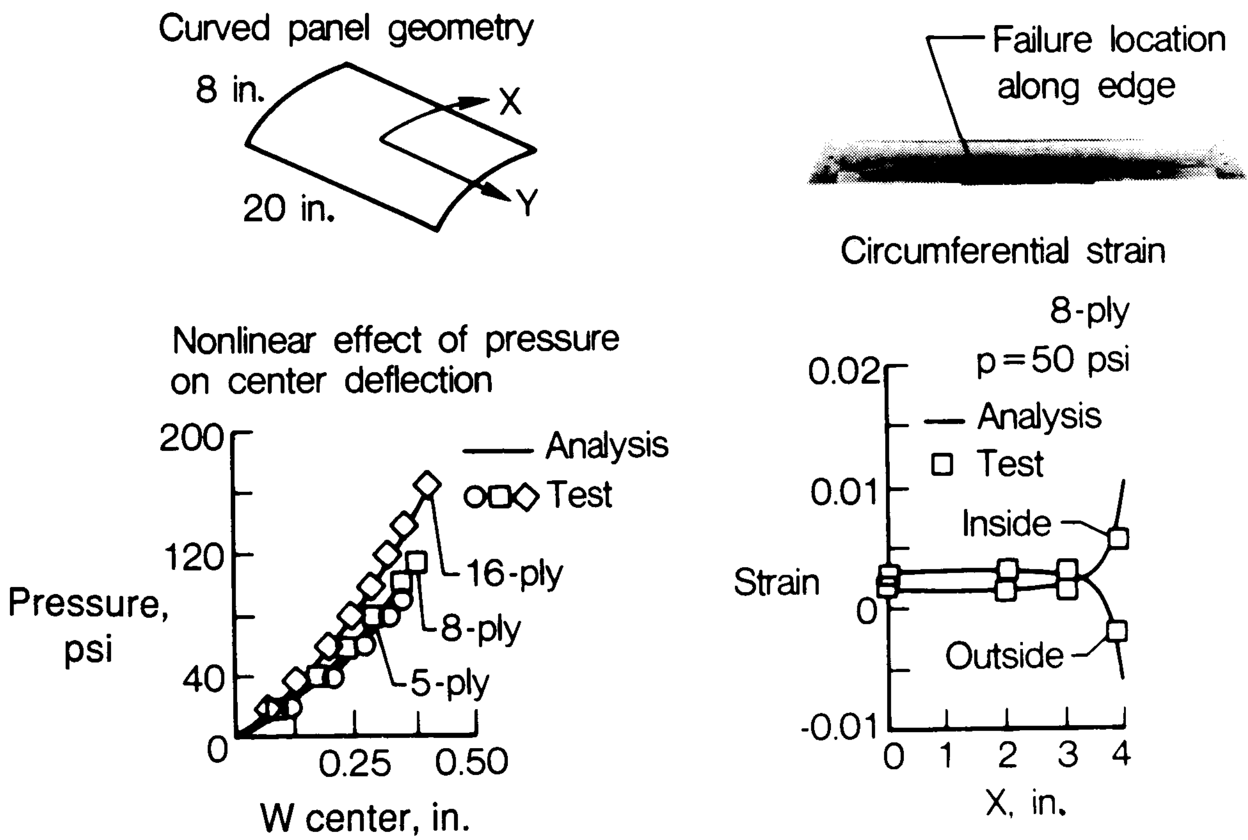


Figure 6

FIRST MAJOR FRACTURE EVENT AND ULTIMATE FAILURE LOAD

Failure results for 4-, 5-, 8- and 16-ply-thick graphite-epoxy panels loaded by internal pressure are taken from reference 1 and are shown in figure 7. The graphite-epoxy panels are made from $[\pm45]_S$, $[\pm45/90]_S$ and $[\pm45]_{2S}$ laminates and $[\pm45/0/90]_S$ and $[\pm45/0/90]_{2S}$ quasi-isotropic laminates. Results are also shown for 0.020- and 0.040-inch-thick aluminum panels for comparison. Strain gage data from back-to-back strain gages near the edge of a panel where the bending strain gradient is severe (see figure 6) are shown in the upper right figure as pressure is increased and the results indicate that local failure can occur in this region before ultimate failure. The ultimate failure pressures of the graphite-epoxy panels are not a linear function of panel thickness. All failure events for the graphite-epoxy panels occurred above 50 psi of internal pressure which is well above the operating pressure of a transport fuselage.

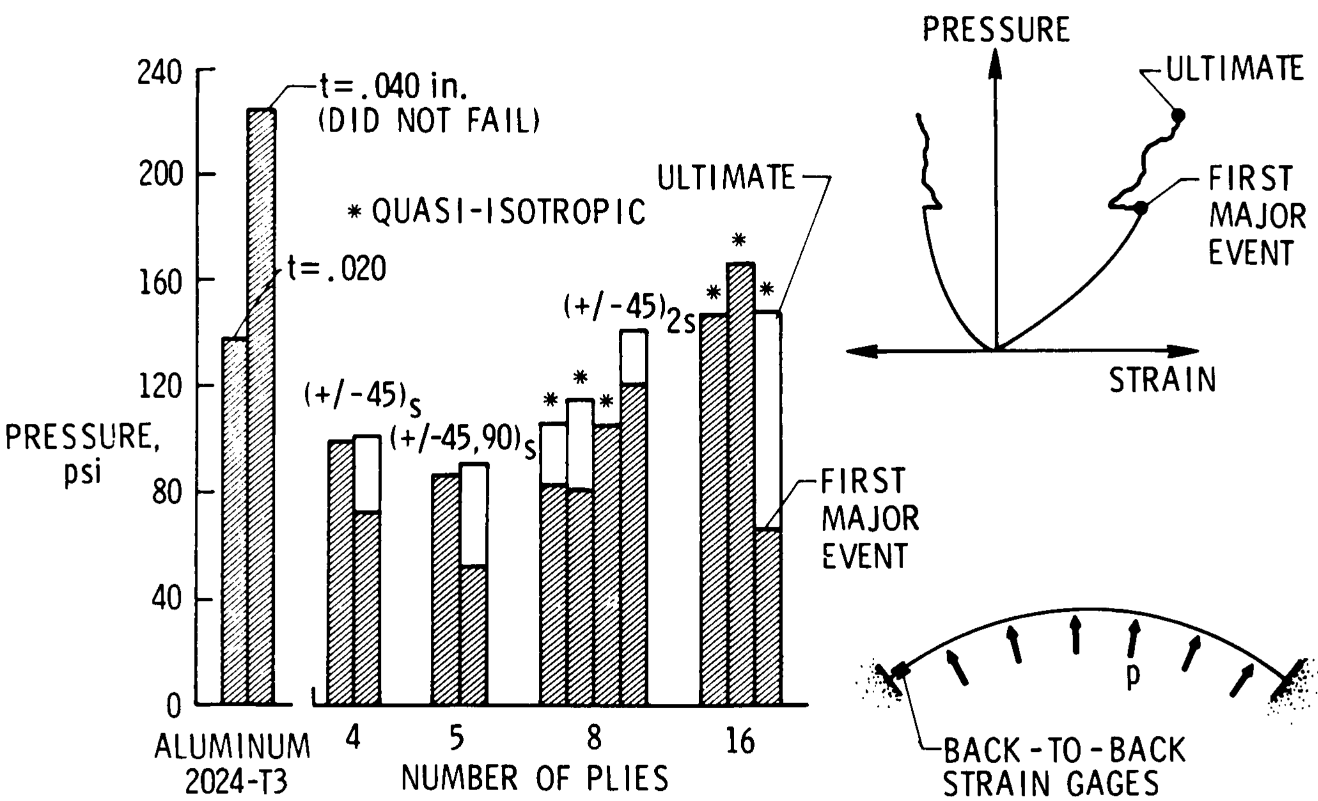


Figure 7

EFFECT OF STIFFENER BENDING STIFFNESS ON PRESSURIZED GRAPHITE-EPOXY PANEL RESPONSE

The influence of stiffener bending stiffness on the response of a stiffened graphite-epoxy panel subjected to pressure loading was studied in reference 2 and some results of that study are shown in figure 8. The middle left and upper right figures show the out-of-plane deflection w distribution across the panel at midlength for 14 psi applied pressure. The middle left figure shows the effect of changing the stiffener height on the skin deflections. The shorter stiffener has relatively low bending stiffness and has relatively little effect on the deformation shape of the skin. The taller stiffener has relatively high bending stiffness and causes the skin to deform into a different shape than for the shorter stiffener. The upper right figure shows the effect of changing the stiffener attachment flange bending stiffness on the deformation shape of the skin for the taller stiffener. Increasing the thickness of the stiffener attachment flange changes the shape of the skin deformation near the stiffener. These results indicate that the deformation shapes of stiffened panels can be significantly influenced by the bending stiffnesses of the stiffener. These deformation shapes suggest that the stresses in the skin are also significantly influenced by the stiffener bending stiffnesses. An example of the interface normal stress between the flange and the skin of one of these panels near the edge of the flange was taken from reference 3 and is shown in the lower right figure. These results indicate that the interface stress gradients are influenced by the nonlinear response of the skin.

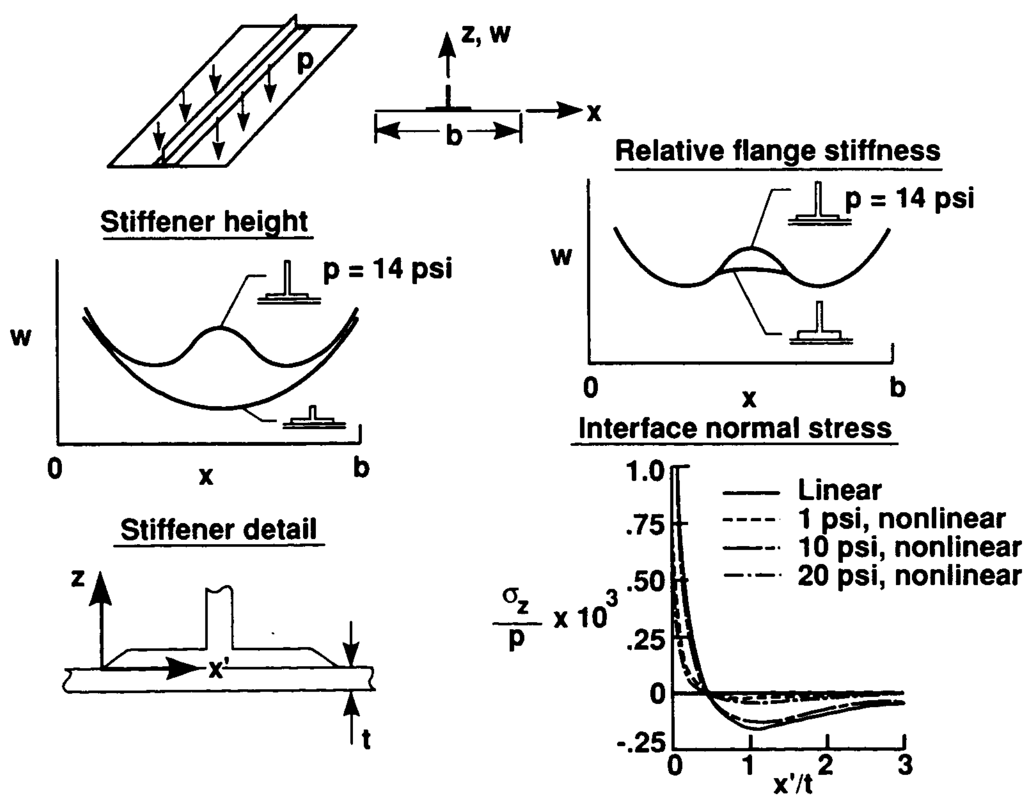


Figure 8

NONLINEAR EFFECTS INFLUENCE BENDING RESPONSE OF COMPOSITE CYLINDERS

The effect of bending loads on graphite-epoxy cylinders is being studied by Mr. Hannes Fuchs under NASA Grant NAG1-343 with VPI and some results of nonlinear analyses from this study for $[\pm45/0/90]_S$ quasi-isotropic cylinder with length-to-radius ratio of 2 and radius-to-thickness ratio of 150 are shown in figure 9. The distribution of the radial deflection w normalized by the shell thickness t for different values of applied bending moment M normalized by the buckling moment M_{cr} are shown in the right-hand figures for the generators with maximum compression and tension stresses. The contour plot shows the radial deflection pattern for the entire shell. These results indicate that the bending in the skin at the ends of the shell grows significantly on the compression side of the shell as the value of M is increased. These results suggest that high values of stresses will occur in this local region of high deformation gradients.

8 ply quasi-isotropic cylinder, L/R=2, R/t=150

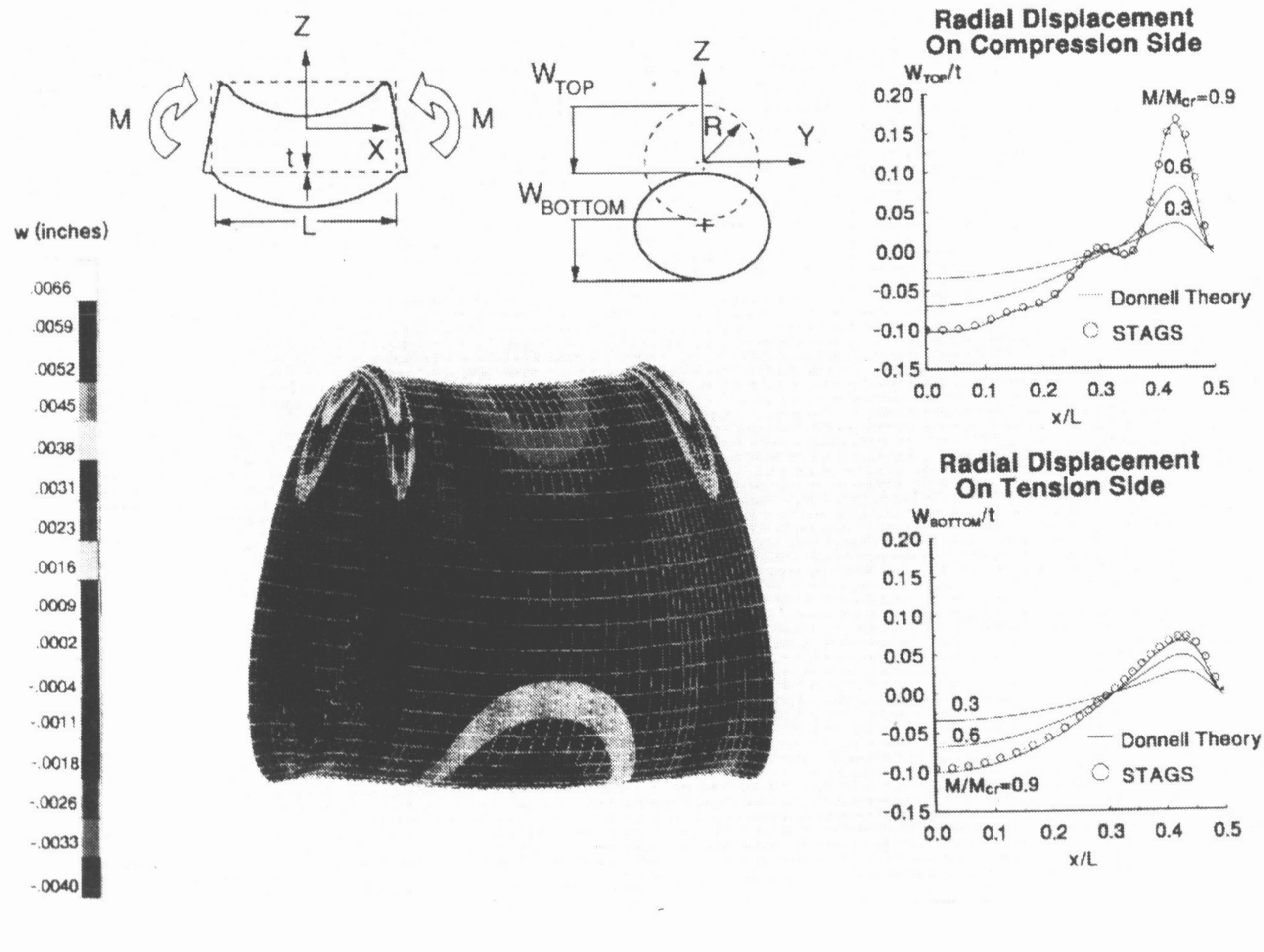
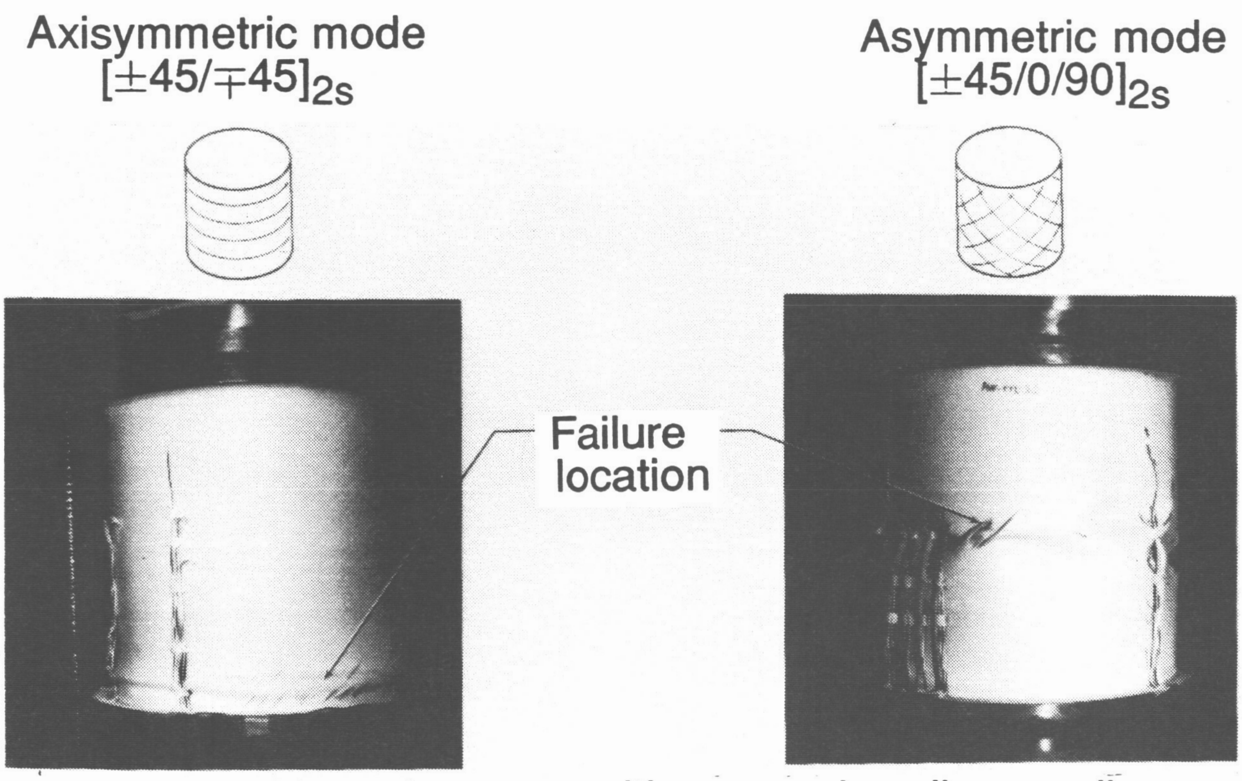


Figure 9

COMPOSITE SHELL FAILURE MODE AFFECTED BY BUCKLING MODE SHAPE

The results of buckling tests for two graphite-epoxy cylindrical shells with different skin laminates are shown in figure 10. These shells are 16 inches long, 0.08 inches thick and have an 8-inch radius. The two shells buckled into different buckling mode shapes; the $[\pm45/\mp45]_{2S}$ shell buckled into an axisymmetric mode shape with larger local bending deformations at the ends and the $[\pm45/0/90]_{2S}$ shell buckled into an asymmetric diamond-pattern mode shape with larger bending deformations along the nodal lines at midlength. Failure occurred for both shells where the local bending deformations were the largest. The results in figures 9 and 10 suggest that regions of a shell with significant local bending deformations should be considered potential failure locations. The results in figures 7 and 8 indicate that significant local bending deformations can also be caused by internal pressure. The combination of internal pressure and mechanical loads with compression and shear components may amplify the local bending deformations in a shell which may affect failure loads and locations.



- Failure initiates in zones with severe bending gradients
  - End bending boundary layer for axisymmetric mode
  - Interior nodal line for asymmetric mode

Figure 10

EFFECT OF CIRCULAR CUTOUTS ON COMPOSITE CYLINDER COMPRESSION STRENGTH

The results of tests for four graphite-epoxy cylindrical shells with 1-inch-diameter cutouts and different skin laminates are shown in figure 11. The shells are 14 inches long, 0.08 inches thick and have an 8-inch radius. For the $[\pm45/0/90]_{2S}$, $[\pm45/0_4/\mp45]_S$ and $[\pm45/90_4/\mp45]_S$ shells, failure occurred at buckling. For the $[\pm45/\mp45]_{2S}$ shell, failure occurred after buckling and at a lower load than the failure loads for the other laminates. Failure was influenced by the cutout for all four shells regardless of the mode shape. The curvature of the shell induces out-of-plane deformation and stress gradients near the cutout which cause interlaminar failures to occur near the cutout. These failures can propagate circumferentially around the shell as shown in the upper left sketch for the $[\pm45/0/90]_{2S}$ shell. Interlaminar failures also occurred near the cutout for the $[\pm45/0_4/\mp45]_S$ shells, but interlaminar failures along the asymmetric buckling modal lines also occurred as shown in the upper right sketch. These results suggest that the local deformation and stress gradients associated with a local discontinuity in the shell, such as a cutout, can significantly influence the response and failure of the shell.

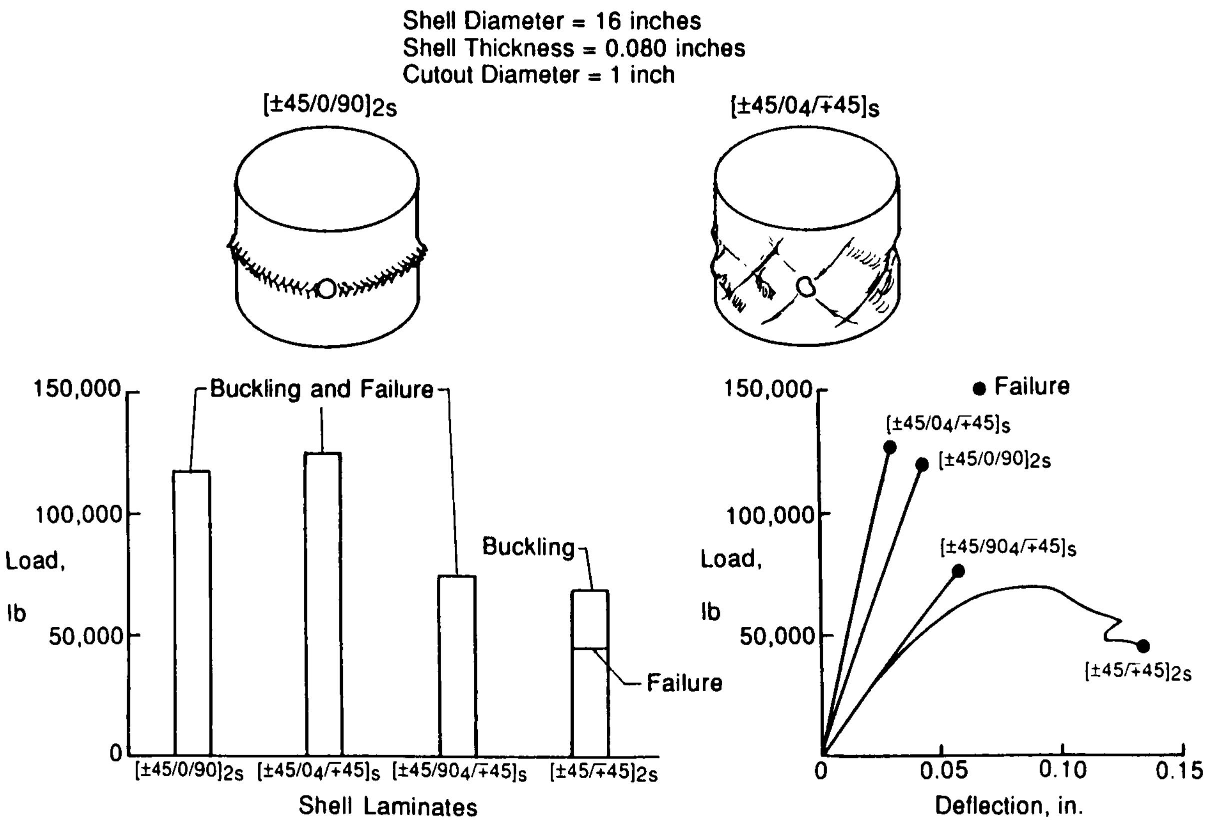


Figure 11

SCALING METHODOLOGY FOR COMPOSITE FUSELAGE SHELL STRUCTURES

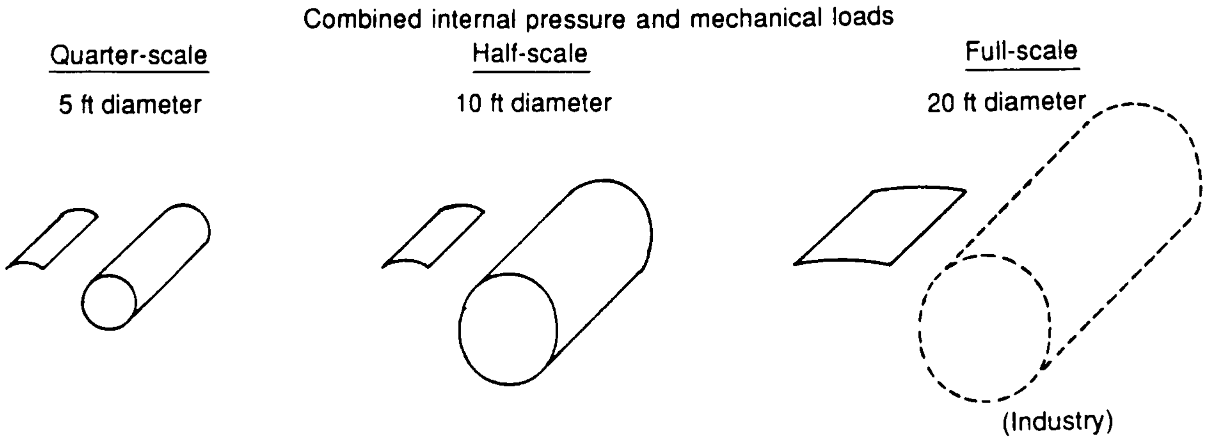
A part of the Langley composite fuselage shell structures research program is the development of structural scaling methodology for composite shells subjected to combined loads. One of the benefits of verified structural scaling methodology includes a reduction in specimen and testing costs during the research and development phases of a new structural design concept. Properly designed subscale models of an advanced design concept should identify some of the critical issues associated with the design before full-scale verification testing is begun. A subscale model, say half or quarter scale, could be used to understand the effects of changing structural parameters on structural behavior. Properly formulated structural scaling methodology should be based on the governing principles of structural mechanics and, as such, should help develop the underlying science and technology base for composite shell structures. Analysis methods verified by testing will be used to formulate the appropriate structural scaling methodology for composite shell structures and parametric studies will be conducted to determine the range of applicability of this structural scaling methodology.

- **Benefits**
  - **Reduced specimen and testing costs during R and D phases**
  - **Improved understanding of parameters that govern structural behavior**
  - **Helps provide underlying science and technology**
- **Scaling methodology based on verified analysis methods and parametric studies**

Figure 12

SCALING METHODOLOGY FOR COMPOSITE FUSELAGE SHELL STRUCTURES

The Langley structural scaling methodology for composite fuselage shell structures will focus on the relationships between full-scale, half-scale and quarter-scale shells with a 20-foot-diameter shell taken as the full-scale shell. Both complete stiffened cylindrical shells and stiffened curved panels will be studied to understand the relationships between complete shells and panels and the effects of changes in geometric parameters on panel and shell behavior. These studies will help to determine what can and cannot be scaled effectively. These studies will also help to identify the interaction between structural parameters, loads and structural response characteristics when geometric parameters are changed. The effect of changing structural scale on failure mechanisms will also be studied. This analysis-based scaling methodology will be verified in the laboratory with test results.



- Determine what can be scaled
- Determine critical failure mechanisms and how they change with scaling
- Determine interaction between structural parameters, loads and structural response mechanisms for scaling methodology
- Verify analysis-based scaling methodology with test results

Figure 13

NONDIMENSIONAL CURVATURE PARAMETER FOR BUCKLING OF ANISOTROPIC SHELLS

An example of analysis-base structural scaling methodology for curved composite panels is shown in figure 14 and is based on the analysis presented in reference 4. The buckling coefficients K_S for a curved panel loaded by a shear stress resultant N_{xy} is shown for an isotropic and an anisotropic panel in the figure as a function of the curvature parameter Z. The parameter Z is a function of the geometric parameters and mechanical properties of the curved panel. The isotropic curvature parameter is a simple function of radius R, width b, thickness t and Poisson's ratio ν as shown in the left equation. The buckling results for an isotropic panel are shown in the left figure and the results indicate that K_S increases as Z increases beyond a value of about 10. The effect of changing any of these parameters can be determined from the curve shown in the lower left figure. The anisotropic curvature parameter is a function of radius R, width b and the membrane and bending stiffnesses of the panel laminate as indicated in the right equation. The buckling results for a $[(\pm 45)_N]_S$ family of composite panels is shown in the right figure. These results indicate than thinner composite panels (i.e., those with low values of N) are affected by the anisotropy of the panel, and buckling results depend on the direction of the applied shear load relative to the panel coordinate axes. Anisotropic coupling can increase or decrease the panel buckling load depending on the direction of the applied load as indicated by the dashed curves in the right-hand figure. Thicker panels (i.e., larger values of N) approach orthotropic panel behavior, and the buckling results are represented by the single solid curve in the right-hand figure.

Isotropic

$$Z = \frac{b^2}{Rt}\sqrt{1-\nu^2}$$

Anisotropic

$$Z = \frac{b^2}{R}\sqrt{\frac{(A_{11}A_{22} - A_{12}^2)A_{66} - A_{11}A_{26}^2 - A_{22}A_{16}^2 + 2A_{12}A_{16}A_{26}}{12\sqrt{(A_{11}A_{66} - A_{16}^2)(A_{22}A_{66} - A_{26}^2)D_{11}D_{22}}}}$$

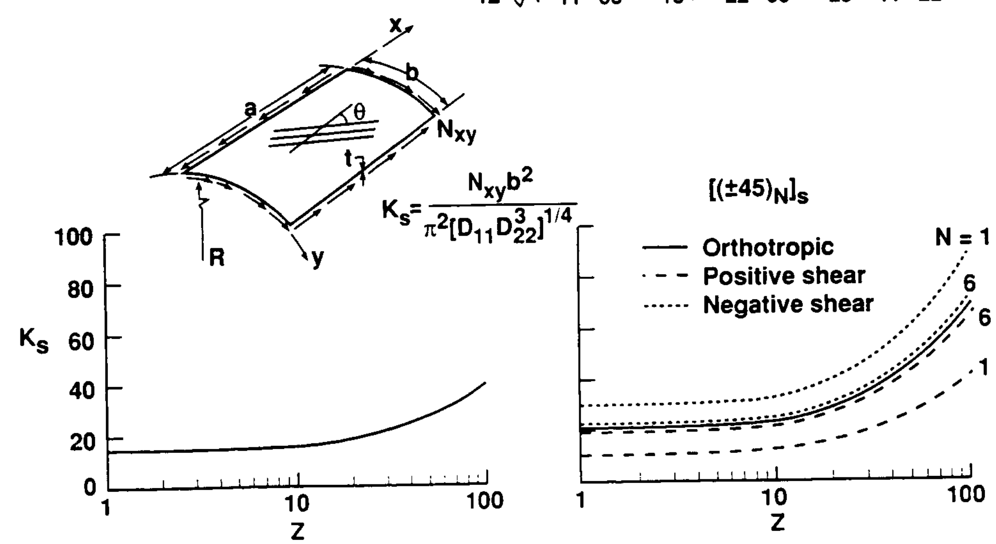


Figure 14

COMPOSITE FUSELAGE SHELL STRUCTURES EXPERIMENTS

Experiments will be conducted as part of the Langley composite fuselage shell structures program to understand the response and failure characteristics of stiffened panels, stiffened shells and structural elements for the panels and shells. Full-scale technology benchmark curved stiffened panels from the Langley Advanced Composites Technology (ACT) program will be tested to verify the behavior of candidate shell design concepts and half-scale pathfinder stiffened shells will be subjected to combined internal pressure and mechanical loads to identify and verify shell behavioral characteristics that cannot be studied at a panel level. These experiments will also be used to verify structural scaling methodology for composite shell structures.

- **Experiments to understand response and failure of stiffened shells, panels and elements**
- **Benchmark curved stiffened panels**
- **Pathfinder stiffened shell structures**
- **Experiments to verify scaling methodology**

Figure 15

PRESSURE BOX

Stiffened panels subjected to combined hoop and axial loads will be tested in the pressure-box fixture shown schematically in figure 16. Internal pressure will be applied to the panel which will generate hoop stress reactions where the panel is attached to the fixture. Hydraulic actuators will be used to generate the axial stresses.

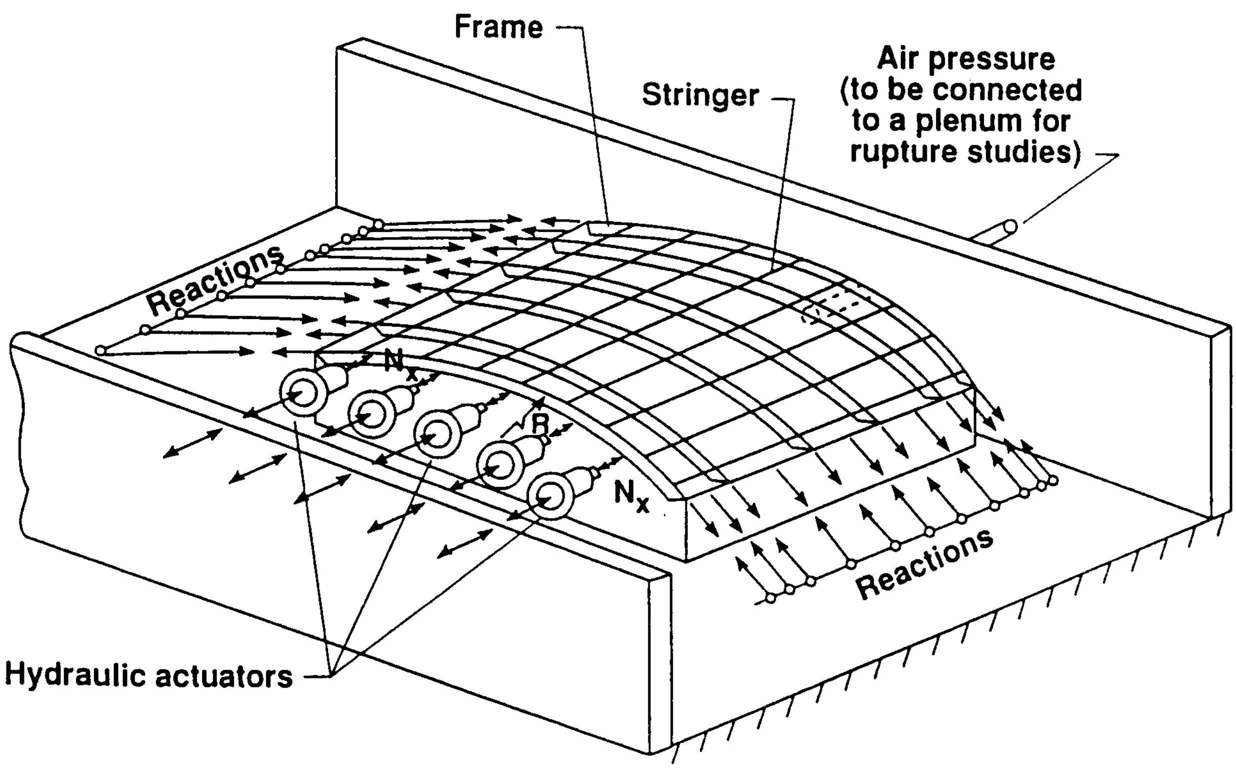


Figure 16

CYLINDER TEST APPARATUS

Stiffened panels and shells subjected to combined loads with a shear component will be tested in the cylinder test apparatus shown schematically in figure 17. A closed-cell test section will be mounted to a rigid backstop at Langley with load introduction adaptor fixtures between the test specimen and the backstop and loading platen. Hydraulic jacks will be used to apply axial, bending, torsion and vertical shear loads to the load platen. Internal pressure will be applied using hydraulic and pneumatic pressure as appropriate.

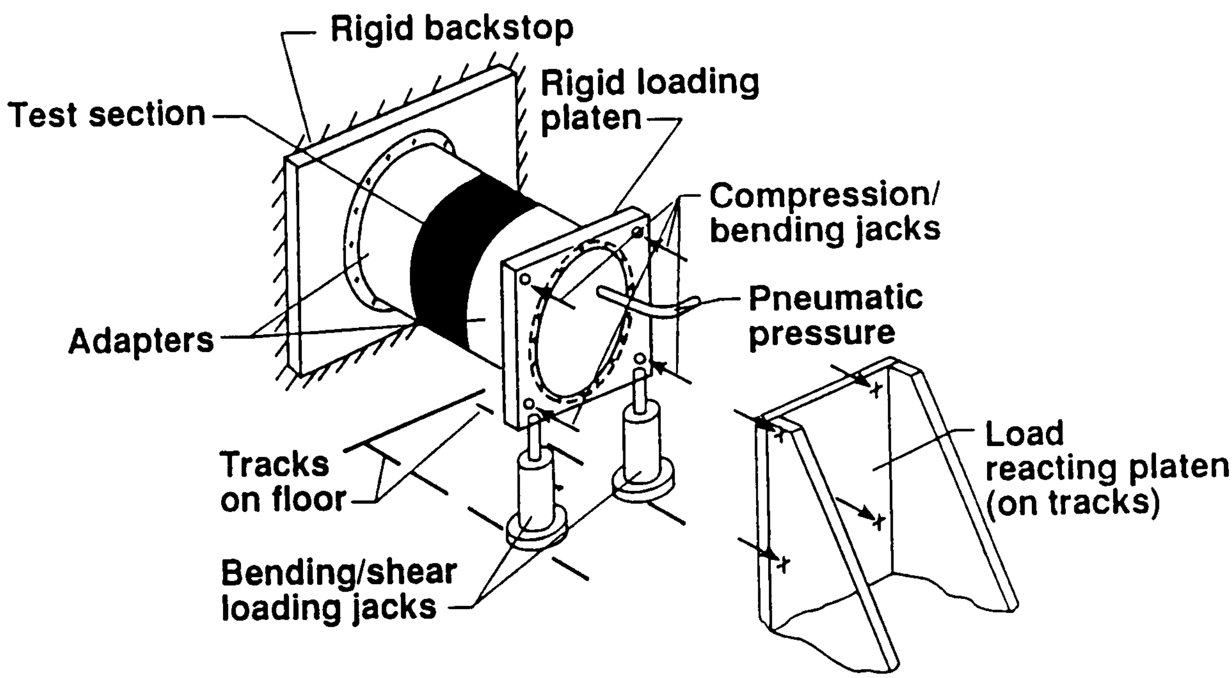
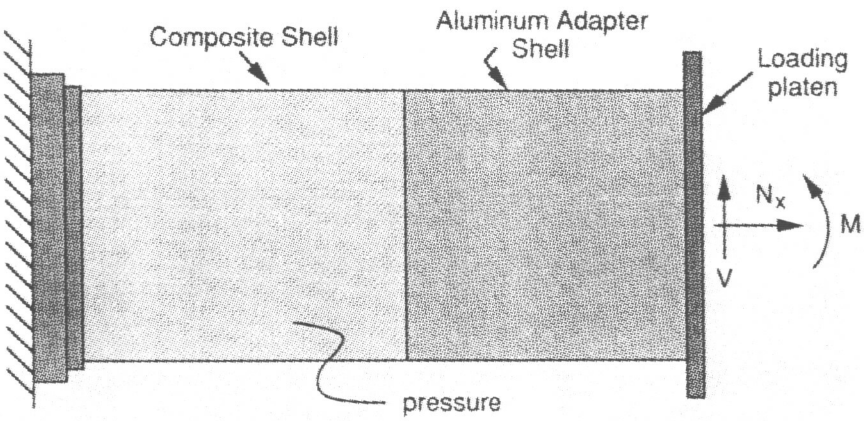


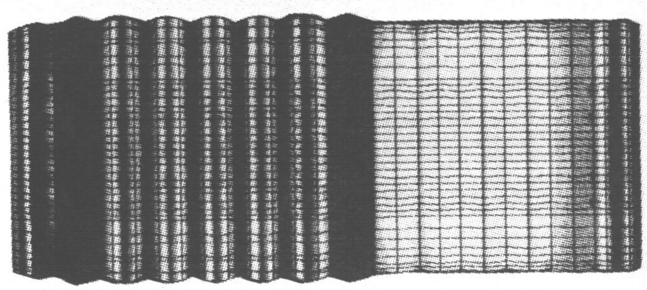
Figure 17

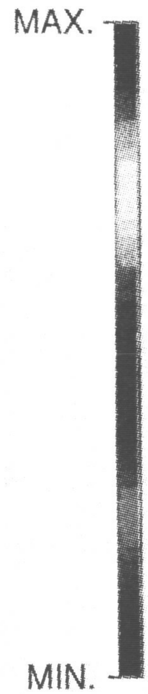
ANALYSIS OF COMPOSITE FUSELAGE SHELL TEST

An aluminum load-introduction adaptor shell is currently being designed and some analytical results for a composite shell loaded by internal pressure and axial tensile loads are shown in figure 18. Attention is being focused on the interaction between the composite test specimen and the aluminum load-introduction adaptor shell to assure that the composite shell behavior is what is expected and that no premature failures at the interface between the composite and aluminum shells occur. The geometrically nonlinear behavior of the composite shell specimen is being considered in the design of the aluminum adaptor shell.



Nonlinear Response to Internal Pressure and Tensile Axial Loads
$[\pm45/90/0/\pm60/\overline{90}]_S$ Shell Laminate





Resultant Hoop Stress

Figure 18

D-BOX FOR CURVED PANEL TESTS

Large-scale curved stiffened panels subjected to combined loads with a shear component will be tested in the closed-cell D-shaped box fixture shown schematically in figure 19. The test panel will be attached to a larger load-introduction or "dummy" panel with the same radius of the test specimen. Analytical studies of the test panel and load-introduction panel configuration will be conducted to quantify the test panel loading including the shear stress resultant N_{xy} and the normal stress resultants and loads in the skin, axial stiffeners and frames N_{skin}, N_S, and N_f, respectively.

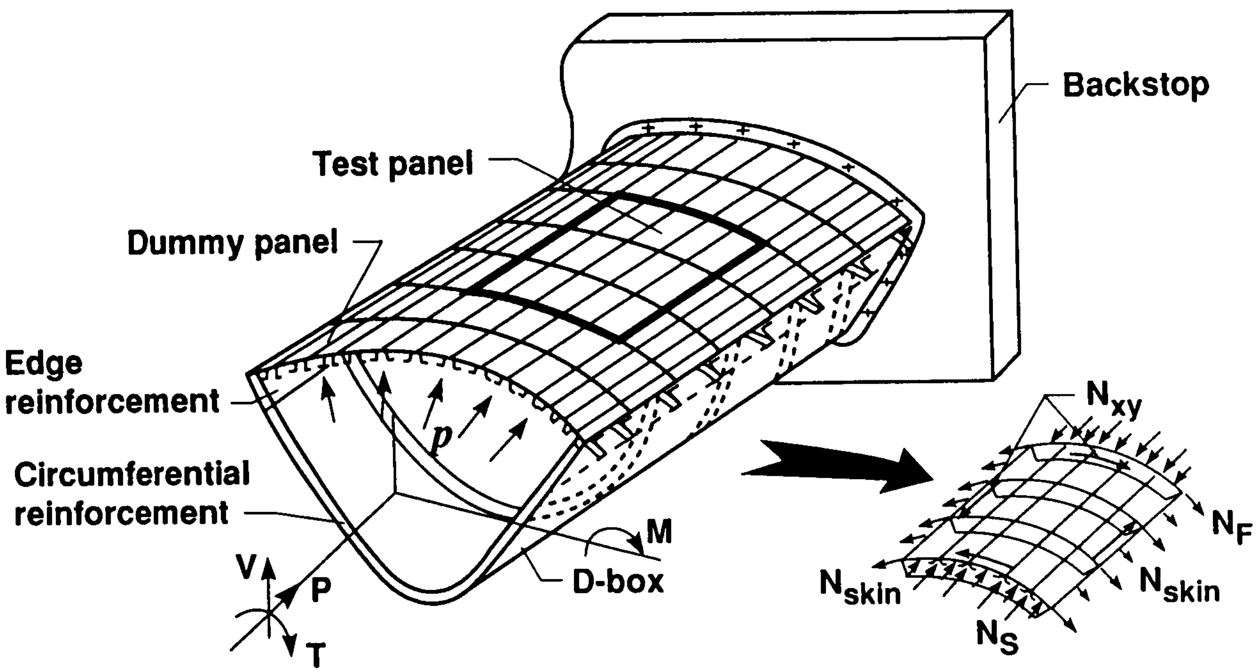


Figure 19

DAMAGE TOLERANCE AND PRESSURE CONTAINMENT FOR THIN-WALLED COMPOSITE SHELL STRUCTURES

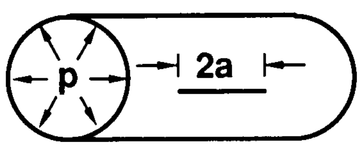
Damage tolerance studies in the Langley composite fuselage shell structures research program will focus on low-energy impact damage and crack growth issues and a limited assessment of high-energy impact damage issues will also be conducted. For low-energy impact damage, a study is being conducted to determine the level of impact damage necessary to cause leaks to occur in thin-walled pressurized composite shells. Studies will be conducted to determine the residual strength of locally damaged shell structures that are subjected to combined internal pressure and mechanical loads. Damage growth characteristics will be identified for curved stiffened panels and shells to help identify critical damage parameters. Damage containment concepts will be developed and evaluated to help provide safer designs. The results of the studies should help define damage tolerance design criteria for thin-walled shells that leak before they burst. A limited high-energy impact damage study will also be conducted to assess the sensitivity of pressurized composite shell structures to very severe damage conditions.

- **Low-energy impact damage and cracks**
  - **Determine damage necessary to cause leaks in pressurized shells**
  - **Determine residual strength of damaged panels and shells subjected to combined loads**
  - **Determine damage growth characteristics and critical damage parameters**
  - **Evaluate damage containment concepts**
  - **Determine damage-tolerance and leak-before-burst criteria**
- **High-energy impact damage**
  - **Assess sensitivity of pressurized composite shell structures to high-energy impact damage**

Figure 20

EFFECTS OF SLITS ON FAILURE OF COMPOSITE SHELLS SUBJECTED TO INTERNAL PRESSURE

Some results of a study of the effects of damage on the burst strength of 12-inch-diameter graphite-epoxy cylindrical shells are shown in figure 21. These results were obtained by Massachusetts Institute of Technology under NASA grant NAG1-991 and are reported in reference 5. Thirty-inch-long unstiffened cylinders with $[90/0/\pm45]_S$, $[\pm45/0]_S$ or $[\pm45/90]_S$ laminates were pressurized to failure with slits of length 2a machined into the shell at midlength. The figure shows that the burst pressure of the shells decreases as the slit length 2a increases and that laminate stacking sequence affects the burst strength.



- AS4/3501-6 graphite-epoxy
- Diameter: 12 in.
- Length: 30 in.
- Thickness
  0.034 in. (6-ply laminates)
  0.042 in. (8-ply laminate)

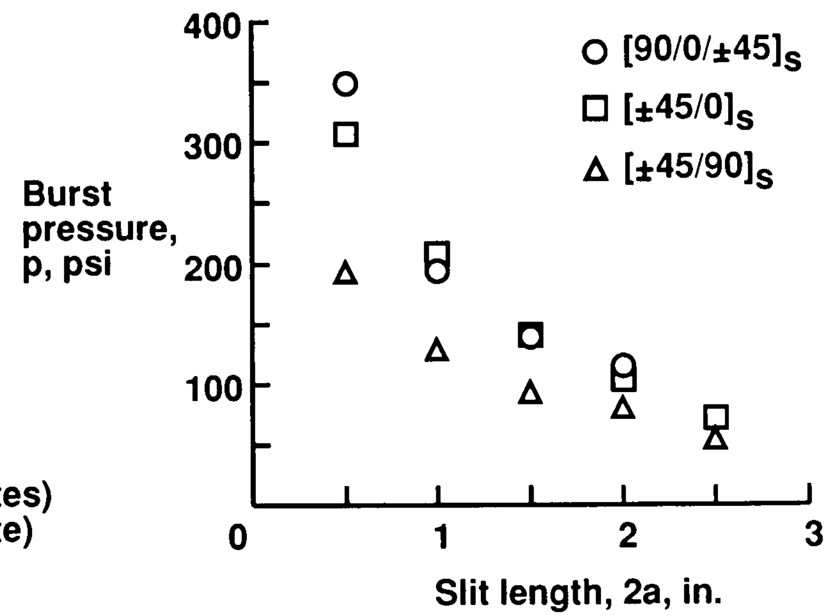


Figure 21

HOOP STRESSES IN COMPOSITE FUSELAGE SHELL WITH DAMAGE

The effect of damage on a full-scale frame and stringer stiffened composite shell subjected to internal pressure and axial tensile loads is shown in figure 22. The shell model is based on the current Boeing design being developed under NASA contract NAS1-18889 and has a 122 inch radius and a 264 inch length and the skin is made from a $[\pm45/90/0/\pm60/90]_S$ graphite-epoxy laminate. The hoop stress resultant distribution for the undamaged shell is shown in the left figure. A 22-inch-long crack was modeled in the skin of the fuselage crown with 11 inches of the crack on either side of the frame at midlength and this frame was also modeled as being broken. The hoop stress resultant distribution for the damaged shell is shown in the right figure. The results indicate that severe hoop stress gradients are present in the vicinity of the damage and the effect of the damage beyond the 2 frames on either side of the crack is shown in detail in the right-hand inset of the right figure. The local bulging of the skin associated with the local radial deflection gradient near the crack is shown in the left-hand inset of the right figure.

Internal Pressure plus Axial Load

Radius = 122 in., Length = 264 in.
Shell Laminate $[\pm45/90/0/\pm60/90]_S$
22 in. Crack in Skin and One Frame

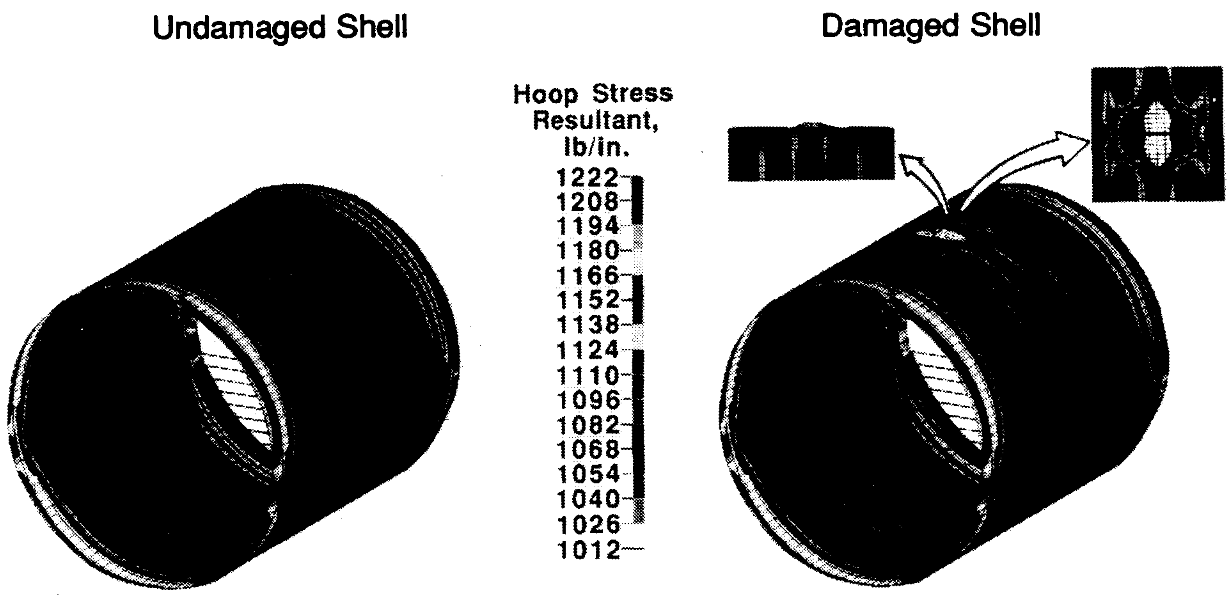


Figure 22

COMPOSITE FUSELAGE SHELL STRUCTURES RESEARCH

The principal activities for the Langley composite fuselage shell structures research program are shown in figure 23 by fiscal year from FY92 to FY95.

Principal Activities by Fiscal Year

FY92

- Develop and evaluate panel and shell concepts and designs
- Analyze response of panels with design details and conduct design studies
- Test panels for effects of discontinuities, impact damage, and internal pressure

FY93

- Conduct nonlinear analyses and design studies for panels and shells
- Test panels subjected to combined loads for response and failure mechanisms
- Analyze response of panels and shells with design details and combined loads
- Test shells for effects of discontinuities, impact damage

FY94-FY95

- Test shells subjected to combined loads for response and failure mechanisms
- Verify damage containment analyses and concepts for pressurized shells
- Verify scaling methodology for panels and shells and conduct nonlinear analyses

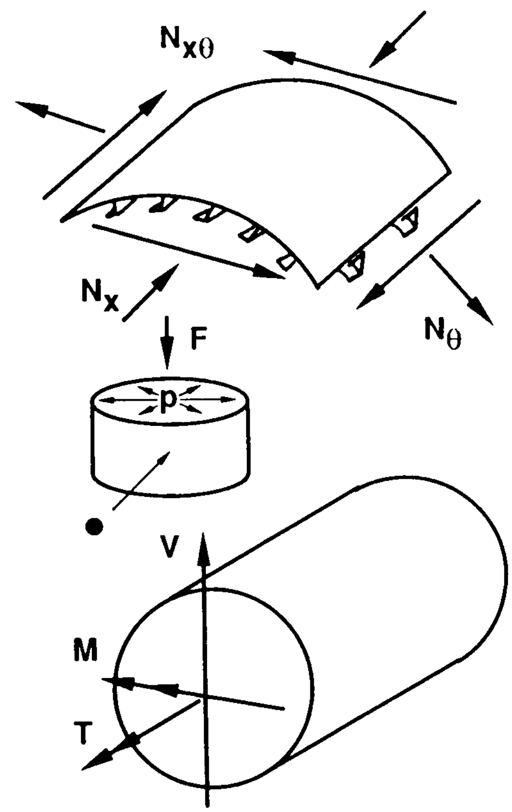


Figure 23

COMPOSITE FUSELAGE SHELL STRUCTURES RESEARCH SCHEDULE

The planned schedule for the Langley composite fuselage shell structures program is shown in figure 24 through fiscal year FY95.

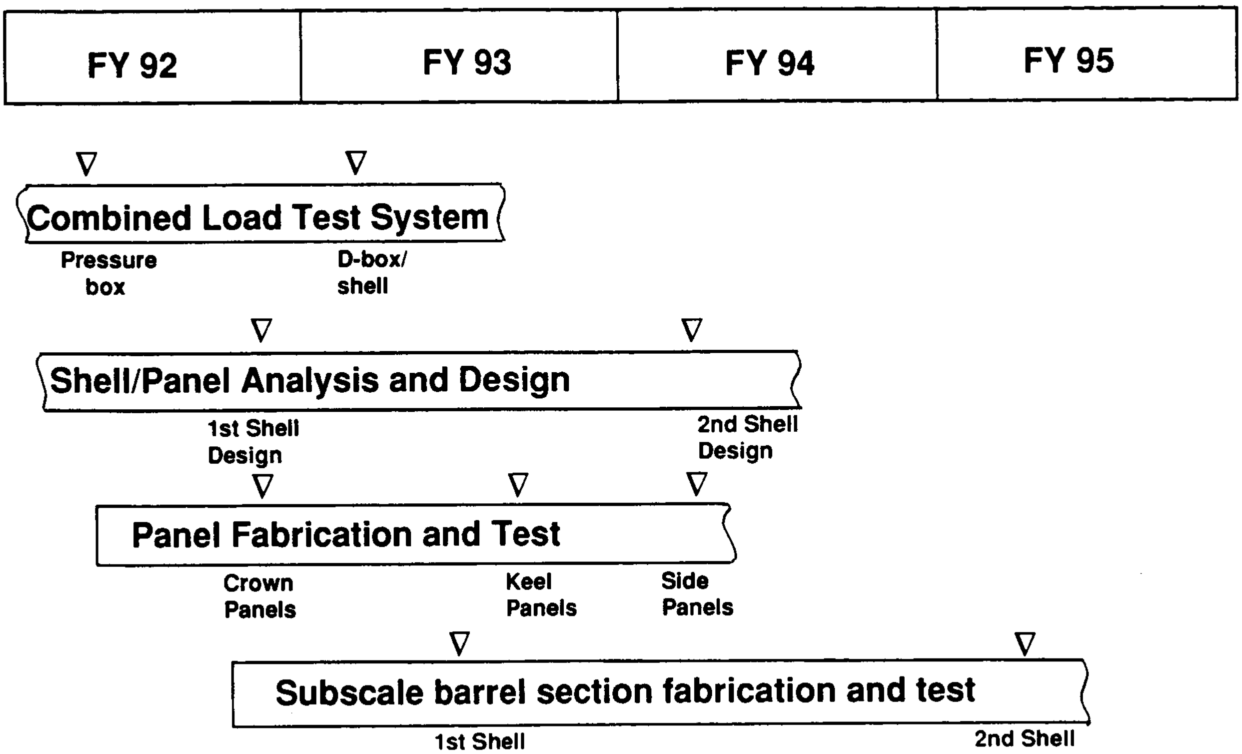


Figure 24

CONCLUDING REMARKS

The composite fuselage shell structures research program at the NASA Langley Research Center will develop verified structural mechanics methodologies for reliably predicting the response and failure of composite frame- and stringer-stiffened shell structures and curved stiffened panels subjected to combined internal pressure and mechanical loads and to local damage. The mechanical loads will include compression, tension, bending, vertical shear and torsional loads. Structural analysis methods that predict the nonlinear response and failure of composite fuselage shell structures subjected to combined loads will be developed and applied to candidate shell designs. Geometrically nonlinear behavior associated with the effects of internal pressure on skin deformation and postbuckling behavior will be included in the analysis and design of candidate shell structures. Structural details, discontinuities and eccentricities that generate local stress and deformation gradients and the interaction between the subcomponents of stiffened shell structure will be studied in the program. Structural sizing procedures that provide minimum-weight designs for stiffened composite fuselage shells subjected to combined loads will be developed and used to conduct parametric studies to determine the sensitivity of the shell behavior to changes in structural parameters. Structural scaling methodology will be developed for composite fuselage shells subjected to combined loads to relate full-scale designs to half-scale and quarter-scale models of these designs. Tests will be conducted on technology benchmark curved stiffened panels and pathfinder stiffened shells to identify critical failure mechanisms, to verify structural analysis methods, and to understand the effects of local gradients and local damage on composite shell behavior. Studies will be conducted to determine the damage tolerance and propagation characteristics and residual strength of damaged composite stiffened shells subjected to combined internal pressure and mechanical loads and damage containment concepts will be explored. The Langley composite fuselage shell structures research program will contribute to the development of the structures technology necessary to develop full-scale pressurized composite stiffened fuselage structures for future transport aircraft.

C-2.

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