# OUT OF PLANE ANALYSIS FOR COMPOSITE STRUCTURES[1]

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Introduction – The ability to predict stresses and failures due to out of plane loads has gained importance as airframe manufacturers begin to use integral (cocured or bonded) composite structures to maximize performance by minimizing weight. Rapid and accurate analysis methods are needed to reduce the amount of testing required to ensure confidence in integral composite structures. The use of postbuckled composite structures will make these analyses even more important. While three dimensional finite element methods can be used to analyze such structures, they require too much time for preliminary structural sizing.

The matrix properties of today's laminates in conjunction with cocured composite construction can produce delaminations under modest levels of out of plane loads. In addition, out of plane loads are only beginning to be considered during the design, development testing, and certification phases of airframe development.

Under a joint Navy/FAA Contract (Reference 1), the problems resulting from out of plane loads were investigated, and ways were presented for avoiding failures that are caused by these loads. A literature search and a survey of industry contacts were used to identify the failure modes and their causes. Simple two dimensional analysis methods were developed to predict the out of plane failure strengths of composite airframe structures. Element test data were used to verify the analyses. Application of these methods to aircraft structure was demonstrated by predicting the ultimate strength of the McDonnell Aircraft High Strain Wingbox test article. The methods and experience from this program were used to compile a set of design guidelines for designers and analysts.

Review of Out of Plane Structural Failures – The difficulties inherent in predicting failures precipitated by out of plane loads are demonstrated in a number of recent test failures of composite structures. A literature search and a survey of industry contacts revealed several examples of these failures: the Lockheed L-1011

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composite vertical fin, test articles from the Northrop Composite Wing/Fuselage Program, the MCAIR AV-8B rudder, the Northrop F-20 horizontal stabilizer, and the Messerschmitt/Grumman Composite Panel Repair Program demonstration component.

Failures due to out of plane stresses were also encountered during development tests of the MCAIR AV-8B horizontal stabilizer. This stabilizer consists of an all carbon/epoxy torque box with separate leading edge, trailing edge, and tip components. Several possible failures resulting from out of plane loads induced by buckling were identified for the torque box, as shown in Figure 1. A combination of out of plane loads and moments caused failures of both the upper and lower covers of the torque box. Failure of the upper panel of the torque box, in excess of 150% design limit load, was attributed to a combination of out of plane loads and moments, caused by buckling, that caused fasteners to pull through the carbon/epoxy closure spar.

The review of out of plane structural failures identified several failures and their causes. These failures are a result of high interlaminar tensile and shear stresses relative to low interlaminar strengths. These high out of plane stresses and subsequent failures result either directly from the application of out of plane loads or indirectly as a result of laminate geometry under in plane loads. Examples of these loadings are

- indirect stresses in laminate corner radii

- indirect stresses due to thickness changes

- indirect stresses due to panel buckling deformations

- direct stresses due to fuel pressure loads

- indirect stresses due to irregular load paths


These potential loading situations must be examined when designing, analyzing, and certifying composite structures.

Analysis Development and Verification - In Reference 1, methods were developed to minimize the stresses identified above. While these techniques can be used to address a great variety of out of plane loading conditions, they were primarily derived to address the failure sources summarized in Figure 2.

The Curved Laminate Analysis (CLA) is a mechanics of materials formulation that can be used to predict the interlaminar stresses in laminate corner radii. Previous curved laminate analyses were based only on the effects of pure bending. CLA extends this capability to include the effects of applied axial, shear, and pressure loads.

In CLA it is assumed that the curved laminate exhibits linear elastic behavior, and plane sections remain plane. Equilibrium conditions for an angular differential element were used to determine the stress state
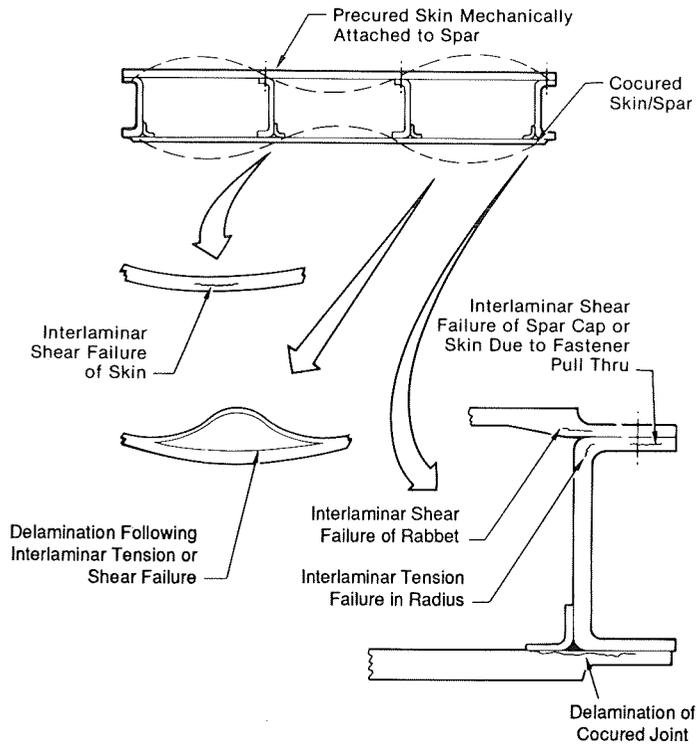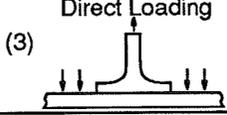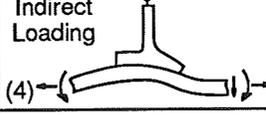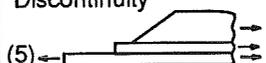
**Figure 1. Possible Postbuckling Related Out-of-Plane Failure Modes in AV-8B Torque Box**

| Problem | Solution Method |
|---|---|
| (1) Load Path Eccentricity | Curved Laminate Analysis (CLA) |
| (2) Variable Thickness Ply Drop-Off | Design Procedure |
| (3) Direct Loading | Elasticity Method (WEBSTER) Interfacial Stress Field Model (ISFM) |
| (4) Indirect Loading | Elasticity Method (WEBSTER) Interfacial Stress Field Model (ISFM) |
| (5) Geometric Discontinuity | Elasticity Method (WEBSTER) |

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**Figure 2. Analysis Methods**

of the curved laminate. CLA loads are defined at the center of curvature, as shown in Figure 3. Stresses are then computed at several user-specified angular positions to find the location that provides the critical stress combination at each ply interface. The Tsai-Hill failure criterion (Reference 2) is used to determine this critical combination of radial and shear stresses.

The Curved Laminate Analysis was evaluated and verified by analyzing carbon/bismaleimide (C/BMI) angle specimen test data. The test specimens were manufactured with a variety of radii and laminate thicknesses and tested to failure under four different environmental conditions. Comparisons of the predicted and actual failure loads are shown in Figure 4.

A fracture mechanics approach (Reference 3) for predicting the initiation and propagation of a delamination in front of a ply drop off was used to conclude that gradual changes in laminate thickness and stiffness generally do not initiate delamination growth. Therefore, out of plane loads due to thickness changes can be minimized and controlled by using established ply drop off design procedures. These design procedures are discussed in more detail in the Design Guidelines section of this paper.

Stresses in integral composite joints due to direct loads (e.g. fuel pressure) or due to indirect loads (induced by out of plane deformations of buckled panels) can be predicted using the Interfacial Stress Field Model (ISFM). This mechanics of materials method models the stiffener flange and the skin as beams, with the cocured or adhesively bonded joint acting as a common elastic foundation, as shown in Figure 5. A set of eight boundary conditions coupled with the governing differential equations of equilibrium establishes a system of eight linearly independent equations that are solved simultaneously. This provides the coefficients that describe the skin and flange deflections. The interfacial normal stress state is determined by the difference of the deflections of the skin and flange at any point along the interface. A shear lag analysis is used to predict the interfacial shear stress state caused by differences in the skin and flange membrane loads. The Tsai-Hill failure criterion is used to couple both the interfacial normal and shear stresses and provide the strength prediction.

ISFM was formulated to analyze the direct and indirect load cases separately by assuming two sets of boundary conditions. A shear load and moment can be applied to the stiffener flange for the direct load case. The buckle deflection needs to be specified for the indirect load case.

The ISFM method was verified for both load cases by correlating the strength predictions with test results. Postbuckling tests of flat composite panels (Reference 4) were used to verify the indirect load case. Figure 6 shows the correlation between the predicted and test separation loads for a range of combined axial and shear load cases.

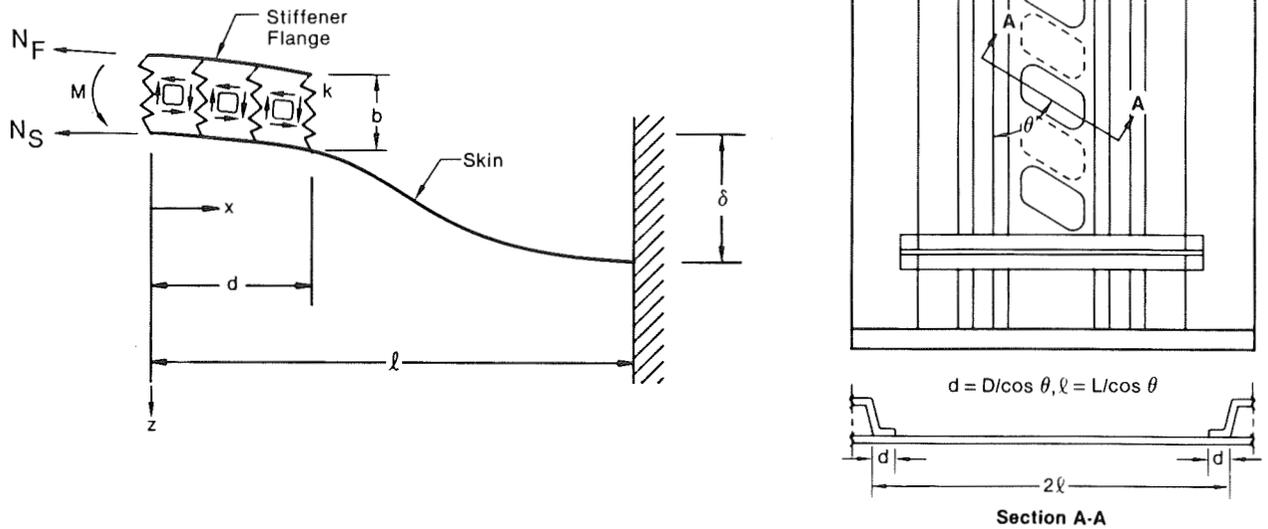The WEB Stiffener Termination (WEBSTER) model offers a more general
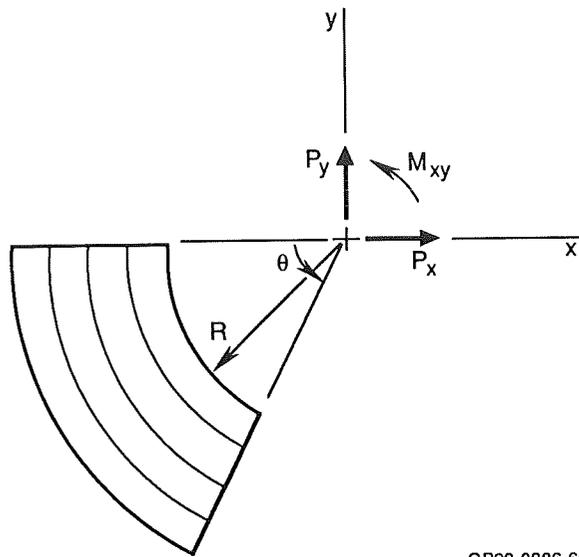
Figure 3. Curved Laminate Analysis Geometry
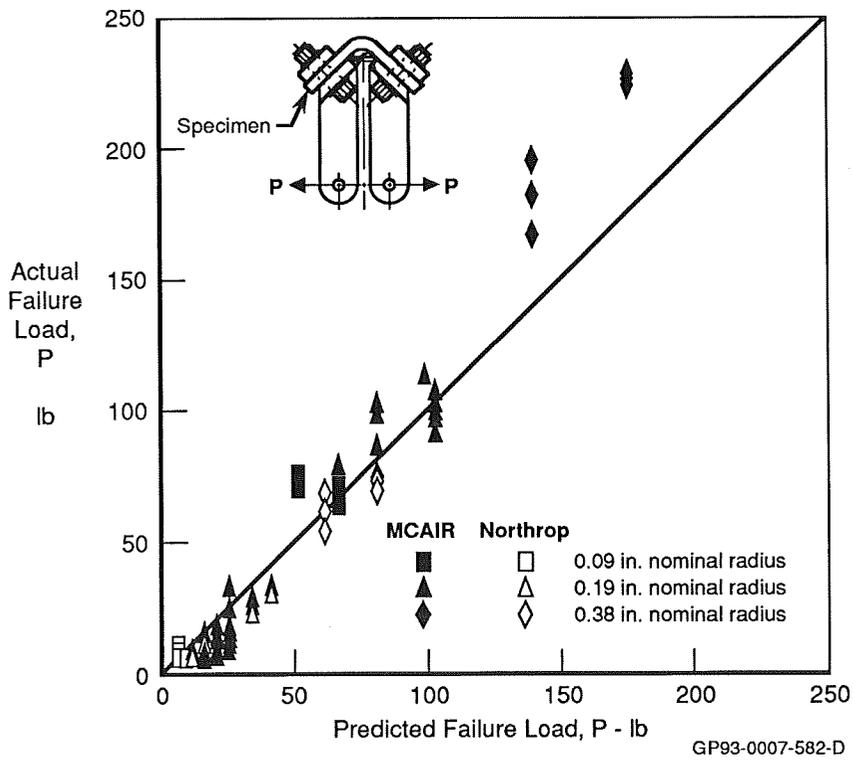


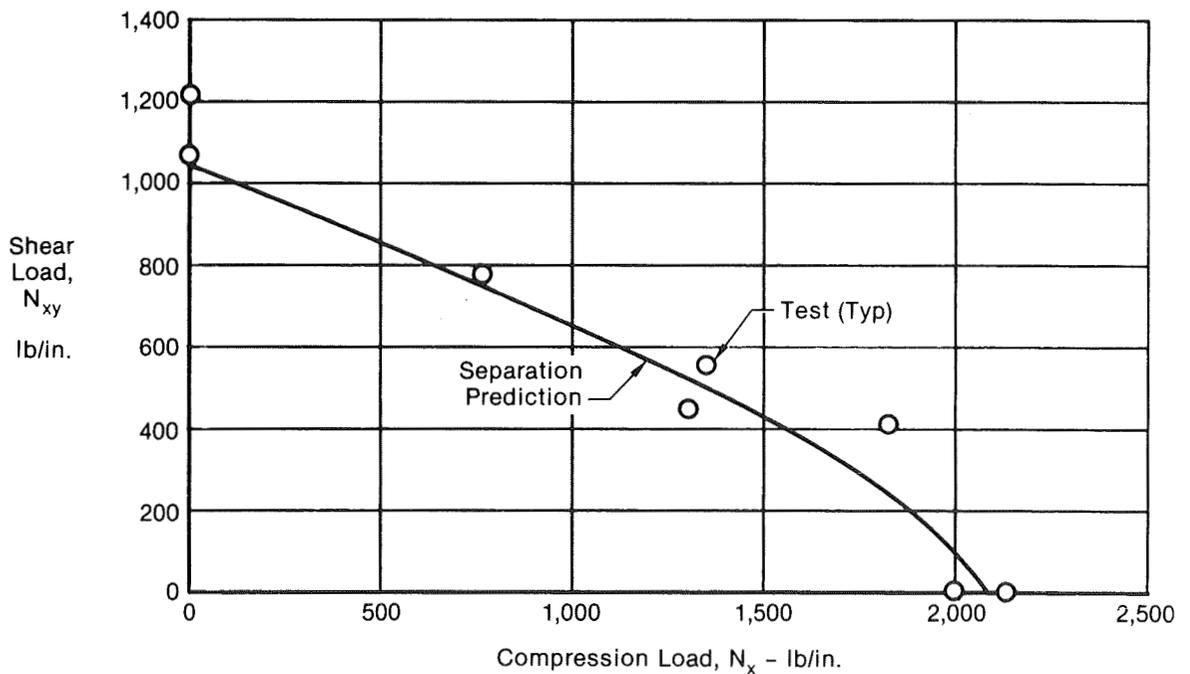Figure 4. Curved Laminate Analysis Verification

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**Figure 5. Interfacial Stress Field Model for Indirect
Load Cases and Its Aplication to a Buckled Panel**



Compression Load, $N_x$ – lb/in.

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**Figure 6. Correlation of ISFM and Test Data of Panels
Subjected to Combined Loads**

method to assess the strength of integral composite joints subjected to direct loads or indirect loads caused by panel buckling. WEBSTER can also be used to predict stresses in composite structures at discontinuities in the geometry, such as the termination of a stiffener or other reinforcement. Figure 7 illustrates the application of the WEBSTER.

In the WEBSTER method the stiffener flange and the adjacent structure (skin) are modeled as separate orthotropic plates that are bonded together with a bondline of zero thickness. The problem is formulated as a generalized plane deformation problem. Boundary stresses are applied to the local region to determine the constants that describe the stress components. The failure function is calculated by averaging the normal and shear stresses over a characteristic distance and applying these values to the Tsai-Hill failure criterion.

The stiffener runout capabilities of WEBSTER were validated using compression panel data. Three specimen configurations were tested to determine the effects of varying the type of stiffener runout. The results of this correlation are shown in Figure 8. Because the Tsai-Hill type failure function is proportional to the square of the applied load, the error in the failure function, shown in Figure 8, is considerably greater than the actual error in the predicted failure load.

An important point to make is that the applicability of each analysis method is dependent on the accuracy of the assumed boundary conditions to the structure being modeled. Figure 9 summarizes the boundary conditions assumed for both the ISFM and the WEBSTER analyses.

Methodology Demonstration - These analyses were used to predict the ultimate strength of the McDonnell Aircraft High Strain Wingbox as a demonstration of the applicability of the techniques to aircraft structural analysis. This structure was a full scale integral composite wingbox, representing that of a next generation fighter structure (Figure 10). Innovative concepts such as integral stiffening and postbuckling were employed to reduce weight and to demonstrate durability and low velocity impact damage tolerance.

An extensive test program was conducted to validate the High Strain Wingbox. Figure 11 summarizes the durability, damage tolerance, and residual strength phases of the test program. Internal fuel pressure and four point bending loads were combined in both static and fatigue tests. Initial static strain surveys were conducted to measure load redistribution due to skin buckling and damage growth. The durability portion of the test program consisted of internal fuel pressure loads and 16000 spectrum fatigue hours (SFH) of bending loads. Nonvisible impact damage and a simulated battle damage repair caused very little degradation of structural response, even after an additional 8000 SFH, for the damage tolerance phase. The residual strength phase of the test program consisted of ultimate bending loads and wing tank pressures for a symmetrical pull up (SPU) maneuver and a rolling pull out (RPO) maneuver. Failure occurred during the RPO condition at the design ultimate load, when the compression cover failed catastrophically. Top and side views of the failure are depicted in Figure 12.
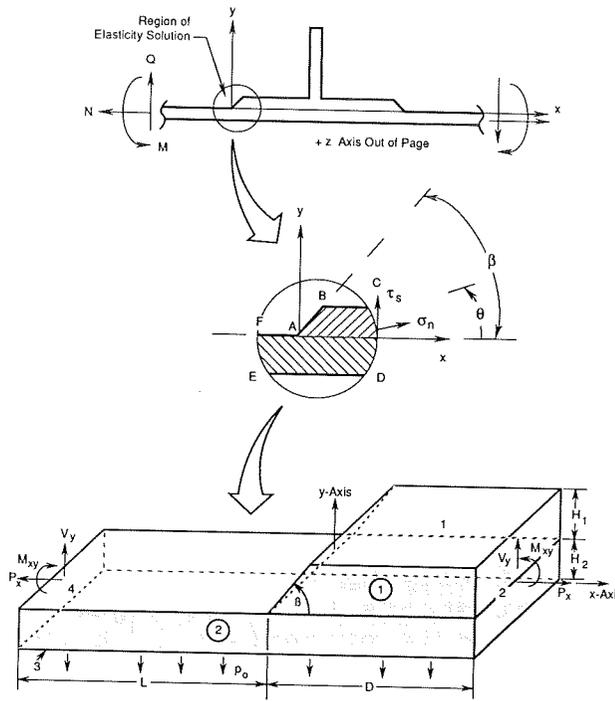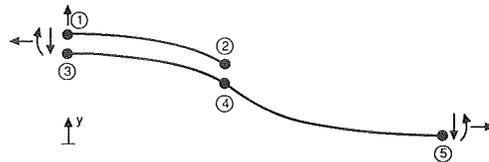
Figure 7.  Local Elasticity Model for Stiffener Termination Analysis



$$F = \sqrt{\frac{F_S{}^2}{F_{Su}{}^2} + \frac{F_T{}^2}{F_{Tu}{}^2}}$$
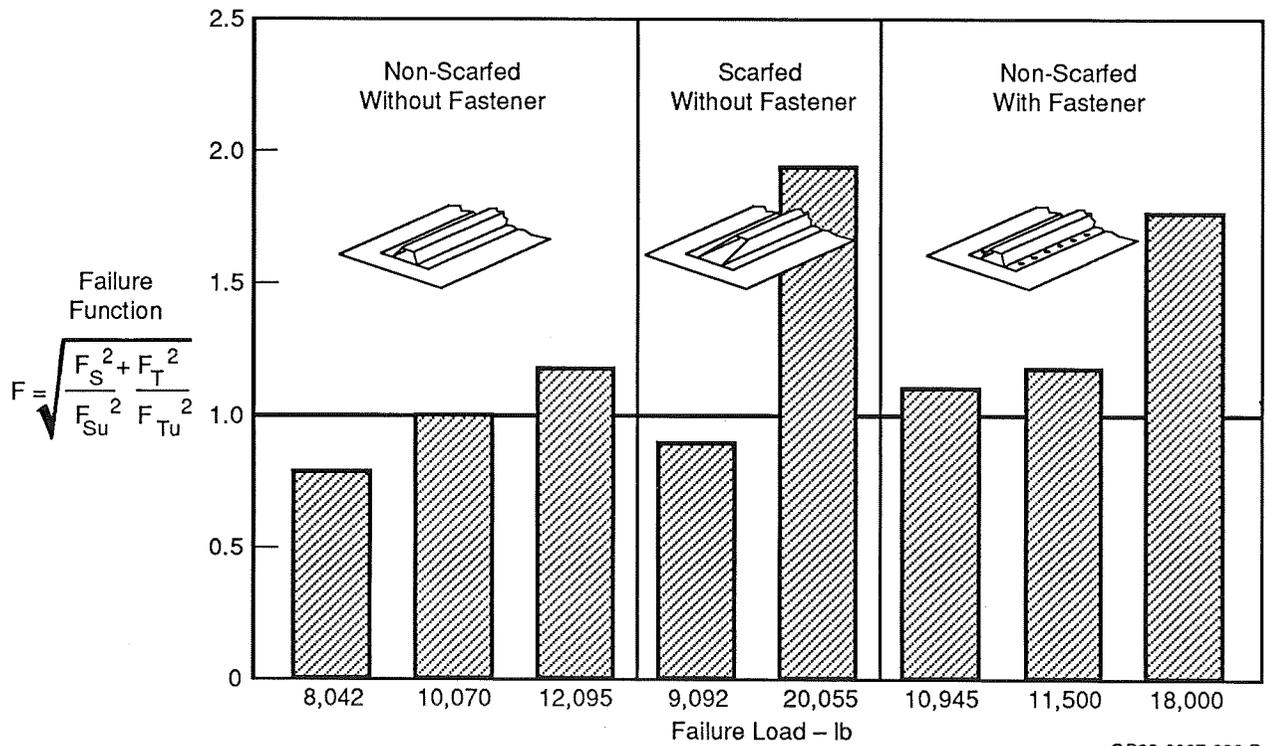
Figure 8.  Compression Hat Stiffener Runout Specimen Failure Correlation

| Location | Boundary Condition | Prescribed Boundary Value | | | | |
|---|---|---|---|---|---|---|
| | | ISFM | | WEBSTER | | |
| | | Direct | Indirect | Case 1 | Case 2 | Case 3 |
| 1 | Deflection | – | δ* | – | – | – |
| | Slope | – | – | – | – | – |
| | Moment | M* | 0 | M** | – | M** |
| | Shear Load | P* | – | V** | P* | V** |
| | Axial Load | – | – | N** | – | N** |
| 2 | Deflection | – | – | – | – | – |
| | Slope | – | – | – | – | – |
| | Moment | 0 | 0 | 0 | 0 | 0 |
| | Shear Load | 0 | 0 | 0 | 0 | 0 |
| | Axial Load | – | – | 0 | 0 | 0 |
| 3 | Deflection | – | – | – | – | – |
| | Slope | 0 | 0 | – | – | – |
| | Moment | – | – | M** | M** | M** |
| | Shear Load | 0 | 0 | V** | – | V** |
| | Axial Load | – | – | N** | – | N** |
| 4 | Deflection | δ** | δ** | – | – | – |
| | Slope | θ** | θ** | – | – | – |
| | Moment | – | – | – | – | – |
| | Shear Load | – | – | – | – | – |
| | Axial Load | – | – | – | – | – |
| 5 | Deflection | 0** | 0** | – | 0 | – |
| | Slope | 0** | 0** | – | 0 | – |
| | Moment | – | – | M* | – | M* |
| | Shear Load | – | – | P* | – | P* |
| | Axial Load | – | – | N* | – | N* |
| Bottom Surface | Pressure | – | – | – | – | p |

\* Denotes applied boundary condition
\*\* Assumed boundary condition - a function of derivation
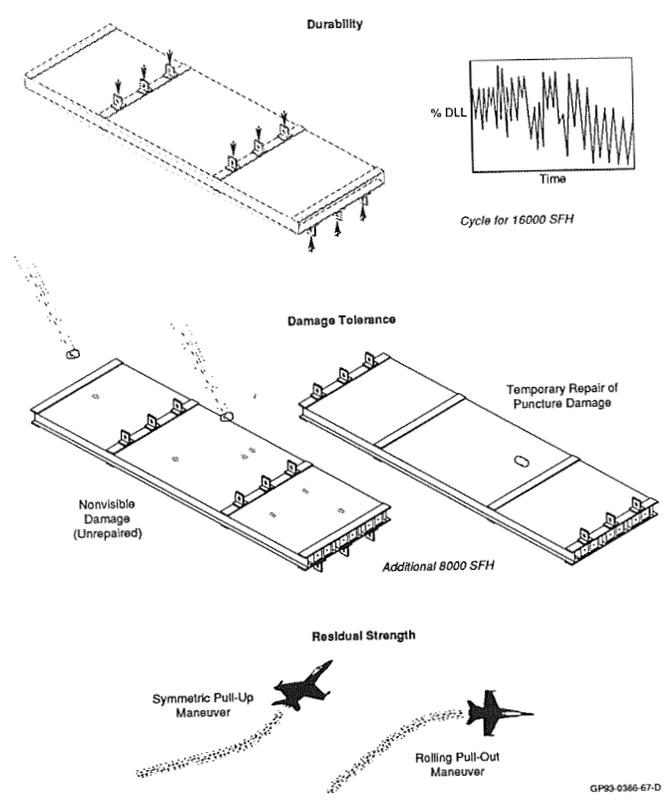
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Figure 9. Summary of Boundary Conditions of Skin/Stiffener Joint Analyses

Figure 9. Summary of Boundary Conditions of Skin/Stiffener Joint Analyses
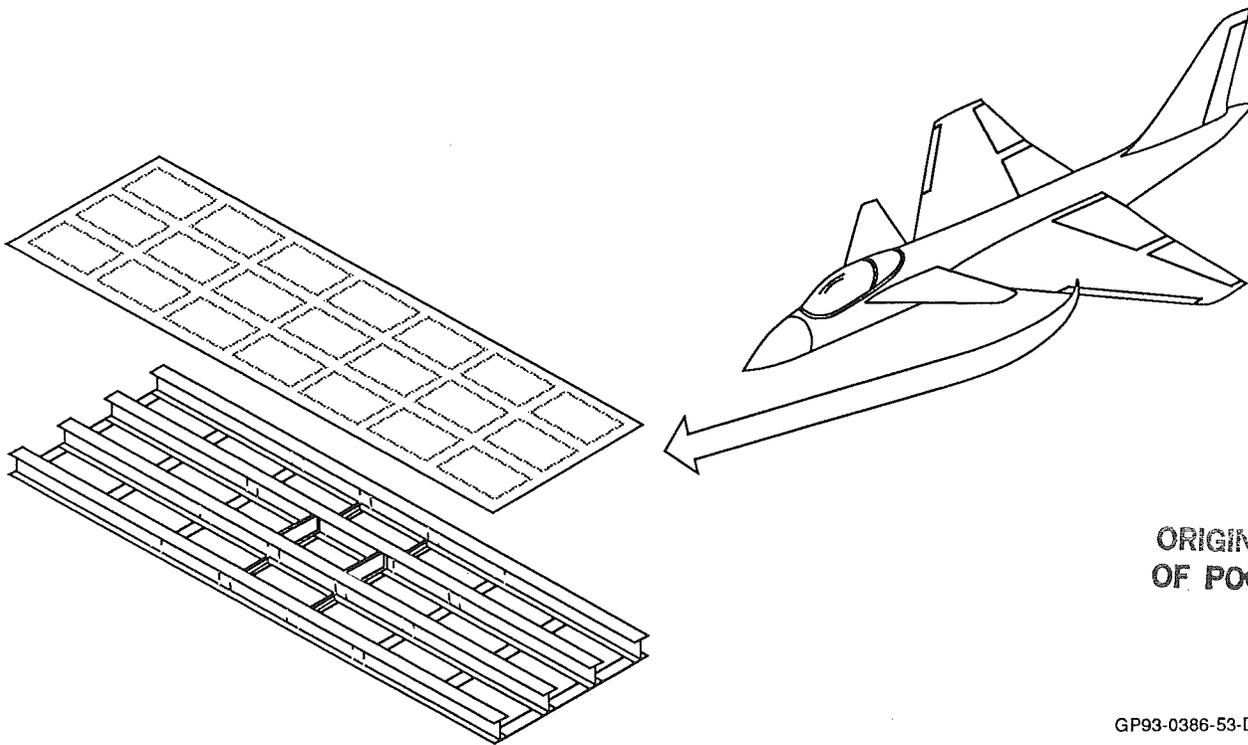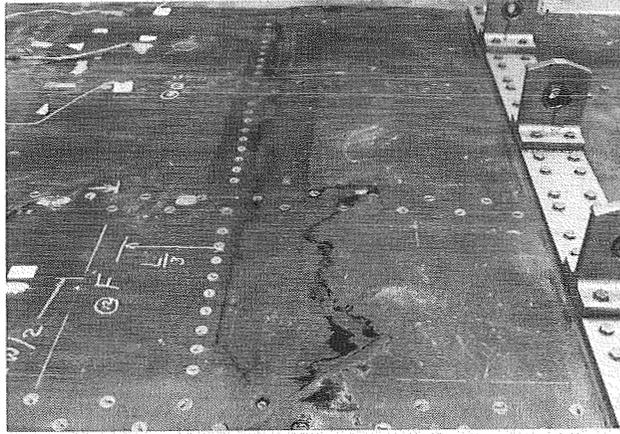
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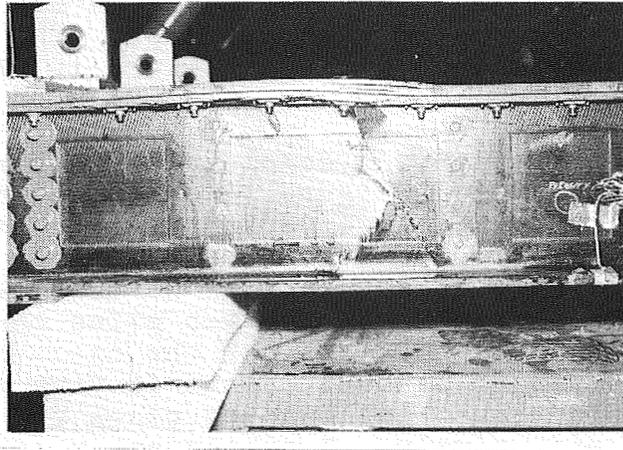Figure 10. MCAIR High Strain Wingbox Concept

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**Figure 11. Test Program for the High Strain Wingbox**

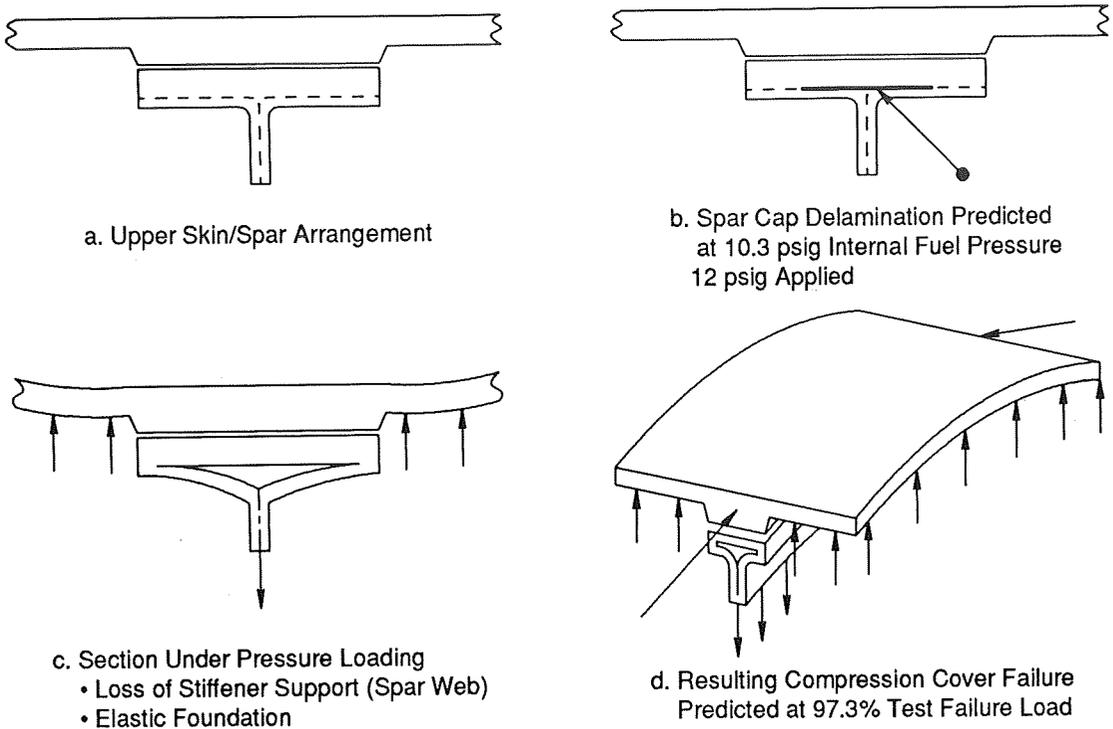a) Across Compression Skin



b) Side View

Figure 12. High Strain Wingbox Failure

The MCAIR High Strain Wingbox was subjected to an extensive structural analysis as part of the methodology demonstration. The purpose of this analysis was to demonstrate that the out of plane methods previously described could have been used to predict the failure of the wingbox, had they been available. These methods could also have been used to improve designs to provide substantially higher strengths.

The analysis of the wingbox was focused on the spar to skin attachments, since the spars provide the reaction for the skin pressure loads and support for postbuckled skins. Failure progression in the wingbox is depicted in Figure 13. The ISFM method was used to predict that the spar cap to web joint would delaminate at internal fuel pressures greater than 10.3 psig. During the durability phase of the test program, a maximum of 12 psig fuel pressure was applied. Therefore, fuel pressure is predicted to cause a delamination in the spar cap. This delamination changed the support of the skin, exchanging a stiff support (the spar web) for a weaker elastic foundation (the delaminated wrapped plies). The reduced stiffness of this joint led to catastrophic failure of the compression cover under wingbox bending loads. The predicted failure load with delaminated spar caps was 97.3% of that measured in test. These analyses show that the box strength was limited by the early failure of the spar caps. The box was predicted to take at least 50% more load without failure if the spar joint does not delaminate.
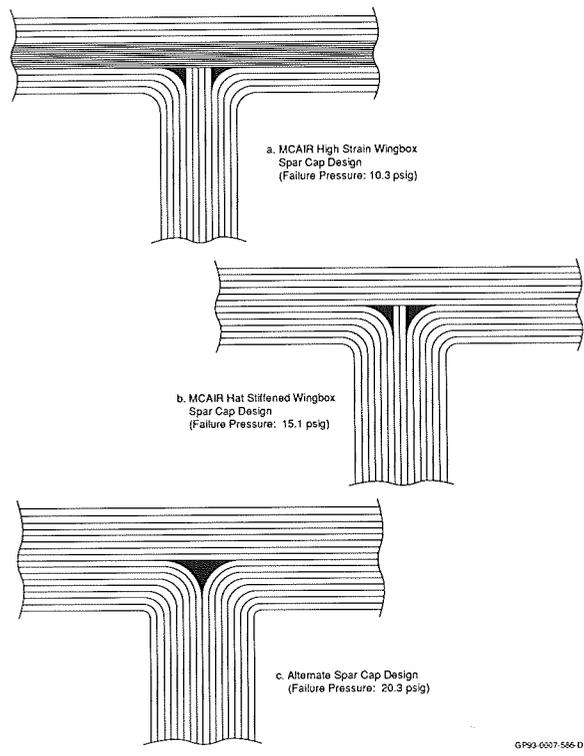
The methodology demonstration was completed by defining an improved spar cap to web joint design. Figure 14 shows the design and the predicted fuel pressure at failure for three of these joints. The design of the joint used in the High Strain Wingbox (Figure 14a) mandated that the pressure load reactions be transferred from the upper skin to the spar webs through both the web plies that wrap into the cap, and the web plies that butt against the cap. MCAIR test data indicates that the strength of butt joints is very low, making them undesirable. Analysis shows that wrapped plies transfer the pressure reaction loads more effectively. A subsequent wingbox, the MCAIR Hat Stiffened Wingbox, featured a spar cap design with fewer plies in the butt joint, and more wrapped plies (Figure 14b). This joint is predicted to fail at 15.1 psig fuel pressure. The third configuration (Figure 14c) represents a spar cap design that features all the web plies wrapped into the spar cap. This joint is strongest and is predicted to fail at 20.3 psig fuel pressure.

The High Strain Wingbox was reanalyzed using CLA, replacing the existing spar cap design with the alternate design shown in Figure 14c. The redesigned spars, and the subsequent lack of delaminations in the spar caps, significantly affect the behavior of the wingbox. The spar caps maintain the stiff support provided by the spar webs, instead of the flexible support provided by the delaminated spar flange. The compressive strength and moment carrying capability of the spar cap/upper skin region is greatly improved for the redesigned wingbox. The predicted load at failure was determined to exceed the RPO maneuver ultimate load by more than 50% when subjected to the same loading conditions.

a. Upper Skin/Spar Arrangement

b. Spar Cap Delamination Predicted
at 10.3 psig Internal Fuel Pressure
12 psig Applied

c. Section Under Pressure Loading
• Loss of Stiffener Support (Spar Web)
• Elastic Foundation

d. Resulting Compression Cover Failure
Predicted at 97.3% Test Failure Load

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**Figure 13. Failure Progression of the High Strain Wingbox**



a. MCAIR High Strain Wingbox
Spar Cap Design
(Failure Pressure: 10.3 psig)

b. MCAIR Hat Stiffened Wingbox
Spar Cap Design
(Failure Pressure: 15.1 psig)

c. Alternate Spar Cap Design
(Failure Pressure: 20.3 psig)

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**Figure 14. Spar Cap Designs**

Design modifications accounting for out of plane loads using the techniques developed in this program would have significantly increased the failure load of the MCAIR High Strain Wingbox. These analysis techniques can be used in the future to provide stronger integrally stiffened composite structures.

Design Guidelines- These methods and the experience gained in this program provide valuable information for the designer/analyst to reduce the risk of out of plane failures of composite parts. The potential for failures due to out of plane stresses can be minimized through proper design, analysis, and test verification. The design guidelines given in Figure 15 will help to further reduce this potential for failure.

As stated previously, out of plane loads caused by thickness changes can be minimized by using established ply drop off design procedures. These have been based on many years of experience at both MCAIR and Northrop, as well as other aircraft companies. The procedures listed in Figure 16 have been proven to produce durable thickness variation in composite structures.

Element test data are required to validate the out of plane analysis methods and to provide data for the determination of stress allowables. Two small elements and one small subcomponent are recommended for this purpose, and are summarized in Figure 17. The curved laminate test specimen (Figure 17a) is similar to those analyzed for verification of the CLA method. This test is recommended for aircraft design development programs whenever curved laminates are used as primary load paths. The laminate layup, thickness, and radius at the corner of the specimen should be designed to satisfy the range of design applications of the specific structure. The test environment should simulate the service environment of the actual structure. A minimum of five specimens should be tested for each specimen configuration and environment.

The stiffener pull off specimen, shown in Figure 17b, is recommended for future tests to verify skin/stiffener separation strength. This configuration is suggested for determining the pull off strength using a failure mode induced by the buckle shape of the skin, rather than direct pull off as has been performed previously. The length and loading conditions of the specimen are selected to match the wavelength of the skin buckle as closely as possible. Previous tests that use transversely cut sections of the stiffener and skin section produce considerable transverse skin bending. This large bending moment at the skin/stiffener joint is not representative of the local loads in actual postbuckled panels.

The stiffener runout specimen is shown schematically in Figure 17c. The purpose of this test is to evaluate designs that attempt to reduce interfacial stresses and to avoid out of plane failures. This test is used to verify the design and WEBSTER predictions. Tests should include a sufficient range of termination angles ($\alpha$) to optimize the structural design. Design features being considered for application to

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• Avoid sudden load path changes within a structure.

• Avoid sudden stiffness changes. Design stiffener runouts so that bending stiffness is continuously runout into the skin.

• If out of plane loads cannot be reduced to acceptable levels, an arrestment mechanism for interfacial failures (fasteners or stitching) should be provided.

• Follow ply drop off procedures in thickness variations (Figure 16).

• Avoid sharp corners (R < 2t) to reduce interlaminar stresses.

• Avoid differences in attachment stiffnesses greater than a factor of 2 in joints.

• Increase skin thicknesses by adding soft plies near discontinuities.

• Optimize skin to stiffener stiffness ratio in postbuckled panels to reduce pull off loads.
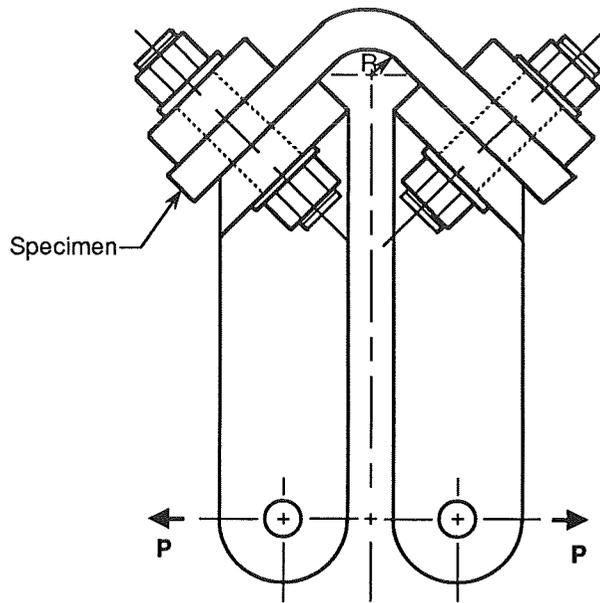
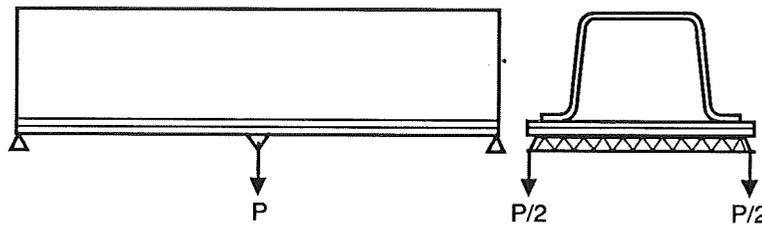GP93-0007-511-D

## Figure 15. Recommended Design Guidelines
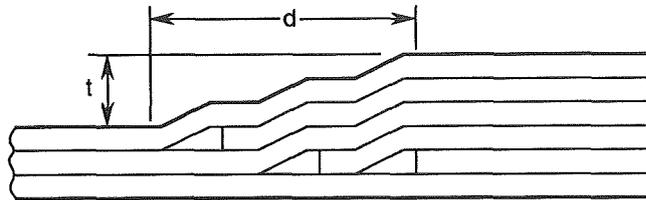


• Maintain symmetry about laminate mid-plane in all constant thickness areas whenever possible

• Maintain d/t > 20 in primary load direction and d/t > 10 in secondary load direction

• Maintain continuous plies on both inner and outer moldline surfaces

• Maintain +45°/–45° ply combination or 45° cloth ply at inner and outer moldline surfaces

• Stack +45° and –45° plies adjacent to each other whenever possible

• Stack 90° plies adjacent to +45°/–45° pairs whenever possible

• Avoid stacking multiple 0° plies together

• Avoid dropping a 0° ply when adjacent to a 90° ply

• Drop off inner plies first

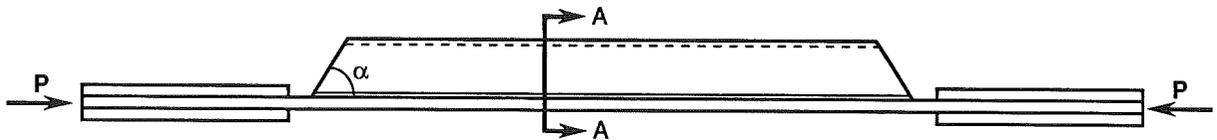• Do not terminate plies at fastener patterns

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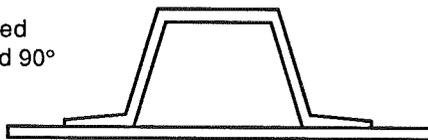## Figure 16. MCAIR/Northrop Ply Drop Off Design Procedures

**a. Curved Laminate Test Configuration**



**b. Stiffener Pull Off Test Configuration**



**A-A** Enlarged
Rotated 90°

**c. Stiffener Runout Test Configuration**

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**Figure 17. Recommended Test Configuration**

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structure must be examined in the design development testing.  One such option is to place "chicken rivets" (mechanical fasteners that provide a redundant load path for cocured or bonded joints) around the termination region such that an out of plane failure can be arrested.  Other options include stitching the stiffener flanges to the skin, scarfing (tapering the stiffener termination) and increasing the skin thickness near the termination site.  Design features like these must be included in the test article.

Summary – Simple two dimensional analysis techniques were developed to aid in the design of strong joints for integrally stiffened/bonded composite structures subjected to out of plane loads.  It was found that most out of plane failures were due to induced stresses arising from rapid changes in load path direction or geometry, induced stresses due to changes in geometry caused by buckling, or direct stresses produced by fuel pressure or bearing loads.  While the analysis techniques were developed to address a great variety of out of plane loading conditions, they were primarily derived to address the conditions described above.

The methods were developed and verified using existing element test data.  The methods were demonstrated using the data from a test failure of a high strain wingbox that was designed, built, and tested under a previous program.  Subsequently, a set of design guidelines were assembled to assist in the design of safe, strong integral composite structures using the analysis techniques developed.

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3.  McCarty, J.E., et al.,"Damage Tolerance of Composites," Interim Reports 1-8, Contract No. F33615-82-C-3213, March  1983 – September 1986.

4.  Ogonowski, J.M. and Sanger, K.B., "Postbuckling of Curved and Flat Stiffened Composite Panels Under Combined Loads," NADC-81097-60, December 1984.