2 General Analysis Practices

The purpose of this chapter is to provide generic guidance on composite structural analysis. It provides discussion and methodology for topics general to analysis of composite aircraft structures, including design guidelines, methods of strength analysis, inspection and test requirements, and preparation of stress analysis reports. This information is for guidance only and the analyst should refer to program documents for specific guidance on composite materials and structures.

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Table 2.1-1—Symbols and Nomenclature

Symbol or Abbreviation	Description	Units
ANOVA	Analysis of variance	
ASIP	Aircraft structural integrity program	
BMI	Bismaleimide	
CAI	Compression strength after impact	in/in
CTE	Coefficient of thermal expansion	
D	Diameter	in
DADT	Durability and Damage Tolerance	
DLL	Design limit load	
DUL	Design ultimate load	
e	Distance from the center of a hole to the edge of a part	in
E	Modulus of elasticity	psi
EOP	End of ply	
$F_{B\text{-}Basis}$	B-Basis allowable stress	psi
FEA	Finite element analysis	
FHC	Filled hole compression	
FHT	Filled hole tension	
FOD	Foreign object debris, foreign object damage	
FV	Fiber volume	
G	Modulus of rigidity (shear modulus)	psi
IDAT	Integrated detail analysis toolset	
IML	Inner mold line	
IRP	Inspection reference panel	
$K_{one\text{-}side}$	One-sided tolerance limit	
M	Applied moment or couple	in-lb
M	Distributed in-plane moments on a panel: M_x, M_y, M_{xy}	in-lb/in
M.S.	Margin of safety	
MRB	Material review board	
N	Distributed in-plane forces on a panel: N_x, N_y, N_{xy}	lb/in
N	Total number of observations in the sample	
NDI	Non-destructive inspection	
OML	Outer mold line	
P	Applied load	lb
P_c	Applied compressive load	lb
P_t	Applied tensile load	lb
P_{xx}	Once per lifetime load, damage tolerance design load	lb
r	Radius of curvature	in
RTM	Resin transfer molding	
s	Fastener spacing	in
s_{dev}	Sample standard deviation	
SPC	Statistical process control	
t	Thickness	in
TAI	Tension strength after impact	in/in
V_{ILS}	Interlaminar shear load	lb
w	Width	in
x_{bar}	Sample mean	
x_i	Value of a statistical observation	

Table 2.1-1—Symbols and Nomenclature

Symbol or Abbreviation	Description	Units
ε	Strain	in/in
τ	Shear stress	psi
τ_{ILS}	Interlaminar shear stress	psi

2.2 Classification of Structure

Every component and structural element of an aircraft has structural requirements. The terminology used in describing the criticality of these requirements is generally customer and specification dependent. Section 2.2 of Reference 2-1 provides a basic overview of these requirements and terminology in the context of metallic structures. That discussion applies directly to composite structures as well and is not repeated here. The discussion in this section highlights the unique implications that these requirements have on composite structures and the reader is directed to Reference 2-1 for the complete discussion on classification of structure. The analyst should refer to the pertinent aircraft design criteria document and specification applicable to the aircraft under consideration.

2.2.1 Structural Criticality

Composite structure is subject to the same requirements and policies identified in Section 2.2 of Reference 2-1 for metallic structures and the full discussion of structural criticality is found in Reference 2-1. This section highlights differences in structural criticality for composite structures. Sources for structural requirements and policies include References 2-13 through 2-18, which are the ASFC Design Handbook, MIL-A-8860, MIL-D-8708C(AS), the Joint Services Specification Guide (JSSG), FAR 25, and FAR 26. Composite structural components have requirements originating from these and other customer- and program-specific sources.

Airframe design requirements for static strength, durability, and damage tolerance assure that structural integrity of composite structures is achieved throughout the product lifecycle. Structural analyses for static strength, durability, and damage tolerance are conducted in like manner as for metallic structures; however, analysis methodologies and allowable properties are tailored to unique failure modes exhibited by composite materials. The nature of analysis performed to address these requirements is summarized below.

- Static strength analysis examines the part's ability to resist a single event loading without failure. Static strength criteria require that a structural member (a) must sustain Design Limit Load (DLL) or 115% DLL without exhibiting detrimental deformation or permanent set and (b) must not rupture or collapse at or below Design Ultimate Load (DUL). The development of strength allowables that correspond to criterion (a) considers load-induced delamination, matrix, and fiber/matrix failures that would result in functional impairment or maintenance action. Generation of strength allowables that correspond to criterion (b) allow local fiber and fiber/matrix failures beyond DLL or 115% DLL. Specific limits on allowable local failure modes are typically defined by program-specific customer requirements.
- Durability analysis examines a part's resistance to damage, including fiber or matrix imperfection, delamination initiation, or delamination growth, for a specified period of time. Durability criteria are primarily addressed by application of empirically derived part thickness values established to preclude nuisance damage and static analysis based on allowable strengths generated considering 4 and 6 ft-lb impact damage.
- Damage tolerance analysis examines a part's resistance to failure resulting from growth of assumed pre-existing damage for a specified period of time. Damage tolerance requirements for composite fracture critical/safety of flight structures are primarily addressed by static analysis methods whose allowable strengths are empirically derived considering high energy impact damage threats. On some programs, damage tolerance requirements are achieved via application of a structural "fail safe" approach, which examines the structure's ability to carry load up to a specified load level even though a portion of the structure has failed or been structurally compromised.

The sections that follow discuss classification designations used with composite structures. Section 2.2.1.1 provides information on structural classification designations and Section 2.2.1.2 discusses classifications used in the Air Force's Aircraft Structural Integrity Program.

2.2.1.1 Structural Classification Designations

Section 2.2.1.1 in Reference 2-1 discusses a classification scheme and terminology that can be used to identify the relative criticality of individual structural components. This terminology provides a clearer understanding of the criticality than the terms "structural", "non-structural" or "secondary structure" and is directly applicable for use with composite structures. The appropriate classification shall be determined by the stress engineer and can be useful in creating a common point of discussion with the design organization.

2.2.1.2 Durability and Damage Tolerance Classification Designations

The Air Force's Aircraft Structural Integrity Program (ASIP) requirements ensure the structural integrity of an aircraft throughout its design life. At LM Aero, ASIP requirements are applied to certification of all composite structures, including structures for non-Air Force programs. Part of the ASIP approach involves the classification of all structural parts based on their criticality, as overviewed in Section 2.2.1.2 of Reference 2-1 and discussed in detail in Reference 2-19.

While the discussion in Reference 2-1is given in the context of metallic structures and is driven primarily by life-based considerations, the same ASIP classification designations apply to composite structure, where the primary consideration is damage initiation and accumulation. The ASIP classification designations, which are discussed in detail in the above references, are as follows:

- Fracture Critical or Damage Tolerance Critical
- Durability Critical or Maintenance Critical
- Mission Critical
- Normal Controls

The decision logic tree for defining and determining structures part classification, and associated documentation; traceability requirements are material independent. Documentation of the approach for air vehicle product lifecycle management is typically a program contract requirement tailored from the Joint Services Specification Guide (JSSG), Reference 2-16, and ASIP requirements, Reference 2-19. Unique aspects of composite structures are recognized in both the design and life cycle management phases.

Durability requirements apply to all composite structural components while damage tolerance requirements apply only to fracture critical/safety of flight structures. While a complete discussion of composite durability and damage tolerance is provided in Section 12.5 of this manual, the discussion in this section outlines the considerations used for composite allowables development.

Durability is the ability of a structural application to retain adequate properties, including strength, stiffness, and environmental resistance, throughout its life to the extent that any deterioration from the delivered condition can be controlled and repaired, if there is a need, by economically acceptable maintenance practices. Each program will coordinate with the customer and/or certifying agency to establish durability criteria for composite parts. Allowables for durability are generally established by test. Programs must demonstrate that composite parts meet the established criteria by analysis and/or test. The delivered condition is a baseline condition which may be far from pristine in nature. Usually there is a requirement to size for initial flaws represented by impacts of 4-6 ft-lbs. The 4 ft-lb impacts are required in areas of low-probability of impact while higher probability impact zones require 6 ft-lbs. Examples of requirements for each zone may be summarized as follows:

For a 4 ft-lb impact zone

- Withstand impact with no visible damage
- Impacts should result in no functional impairment for two design lives
- No failure for design ultimate load including environmental effects

For a 6 ft-lb impact zone

- Probable impact zones are established where this criterion is applied
- Impacts should result in no functional impairment for two design lives
- No failure for design ultimate load including environmental effects

Fatigue loading is typically not a design sizing consideration for composite structural members. Unlike metallic materials, composites loaded in the plane of the reinforcing fibers are generally insensitive to repeated loads when the magnitude is below approximately 65% of the static ultimate strength. Fatigue loads spectra used for durability fatigue tests of composite structural members are typically subject to 50 to 70% load point truncation as a result of this fatigue insensitivity. Fatigue considerations can become a design consideration for composites loaded out-of-plane.

Damage tolerance is the ability of a structure to sustain design loads in the presence of damage caused by fatigue, corrosion, environment, accidental events, and other sources until such damage is detected, through inspections or malfunctions, and repaired. In general, metallic structure is allowed to exhibit a certain amount of sub-critical crack growth within the design life of the component. Detectable cracks are noted and managed as part of the maintenance and inspection process. Composite material life cycles are managed by initial sizing that precludes any incremental growth of defects that can be expected to be present during the manufacturing process or will arise during the normal operation of the structure throughout its design life. Damage threats during life cycle use, usually from impact, that cannot be seen from a normal visual inspection must be considered and must not result in functional impairment for the given inspection interval.

The primary source of composite structure damage is impact. Compression loading, both static and cyclical, after impact is often critical for composite structures. Low-level impact can easily produce non-visible damage and the structure must maintain design load capability until the damage is detected. Tension in damaged composites is not typically an issue but compression in the presence of delaminations can create stability issues because individual fibers buckle in the delaminated area and thus only a portion of the laminate stack is resisting buckling.

Typical requirements for the type and severity of damage to be considered are:

- 4 inch x 0.02 inch deep scratch
- 2 inch circular single planar delamination
- Impact from a 1 inch diameter hemispherical impactor that results in damage barely visible from 5 feet. A maximum cutoff of 100 in-lbs should be used for this requirement. Often, a dent depth criterion is also applied.

Typical sizing considerations are as follows:

- Damage detected during maintenance inspections must withstand a once per lifetime load, P_{xx}, which is applied following repeated service loads occurring during an inspection interval (with appropriate magnification factor). The definition of P_{xx} depends on the certifying authority but might be as follows: once in 20 lifetimes load, maximum limit load, maximum design load, or some percent of limit load.
- All damage that lowers strength below Ultimate Load must be repaired when found.
- Any damaged structure that is repaired must withstand Ultimate Load.
- Structure damaged from an in-flight, discrete source that is evident to the crew[1] must withstand loads that are consistent with continued safe flight.

[1] The damage tolerance design load, P_{xx}, addresses in-flight damage that is not evident to the crew. Such damage that lowers strength below Ultimate Load must be repaired when found.

2.3 Composite Structure Design Guidelines

This section provides guidelines for designing structures with composite materials. The intent of this section is to provide uniformity in general design practices for composite structures. These are only guidelines and good engineering judgment must be exercised in their use.

2.3.1 Minimum Skin Thickness

This section provides guidance on minimum composite skin thickness that can be used with good engineering judgment in lieu of specific program guidance. The examples given below are guidelines that have been used based on specific testing, configurations, and materials and their suitability for a specific application may vary. These guidelines can be used for preliminary design and as a starting point for detailed development work, but programs should execute development test programs to substantiate these minimum thickness values and any deviations from them for certification.

Minimum composite skin thickness is primarily driven by producibility requirements, the need for adequate damage resistance from Foreign Object Damage (FOD) threats, and lightning strike requirements. For all composite structure that functions as a fuel boundary, there is the additional requirement to ensure fuel containment. Consequently, fuel boundary structure has a higher minimum thickness requirement as shown in the requirements given below. Likewise, sandwich facesheet laminates must have sufficient thickness to prevent moisture intrusion into core material where it can become trapped, subjecting the laminate to long term moisture exposure and freeze-thaw cycling.

First, producibility considerations drive minimum laminate thickness. Producibility requires that composite laminates withstand handling, assembly, and maintenance operations. Laminates that are too thin will damage easily from these operations, creating unnecessary waste and cost. Typical minimum thickness values to satisfy producibility requirements are given in Table 2.3-1.

Table 2.3-1—Minimum Thicknesses to Meet Producibility and Fuel Containment Requirements

Location	Sandwich Facesheet (Inches)	Monolithic (Inches)	
		Dry	Fuel Boundary
Exterior	0.024	0.060	0.080
Interior	0.024	0.024	0.040

Next, FOD threats place additional minimum thickness requirements on composite laminates. FOD threats can include bird, hail, runway debris, dropped tools, and other objects as defined in Reference 2-16. Requirements are program specific and driven primarily by customer requirements for airframe maintainability. These requirements are typically specified according to zones of varying probability and severity of impact. Table 2.3-2 provides guidance for composite laminate minimum thickness values to meet FOD and water intrusion requirements.

Table 2.3-2— Minimum Thicknesses to Meet FOD Requirements for Carbon and Glass Reinforced Composite Parts

Material and/or Construction	Low Probability Zone - Closed Out Bays & Internal Areas With Infrequent Maintenance Activity	High Probability Zone - All Other Areas Including All External Surfaces
IM7 Carbon Fiber (BMI & Epoxy Resin)		
(1) Monolithic		
a) Non-fuel Boundary	.042 (tape or fabric)	.064 (tape)
		.066 (CSW)[1]
		.053 (CSW[1]+S-2 fabric) BMI only
b) Fuel Boundary	.064 (tape+ fuel tank coatings required)	.080 (tape)
	.083 (fabric)	.083 (CSW)[1]
(2) Sandwich Facesheet / N636 Non-metallic Core		
a) Non-fuel Boundary	.033 (tape or fabric)	.053 (tape) or .050 (fabric)
b) Fuel Boundary	Not Allowed	Not Allowed
S-2 GLASS (BMI)		
(1) Monolithic	.039	.039
(2) Sandwich Facesheet / N636 Non-metallic Core	.039	.039

[1]CSW—Crowfoot Satin Weave fabric

Finally, lightning strike requirements place minimum thickness requirements on composite laminate skins (with or without core) in areas over fuel, pressurized (e.g., cockpit) zones, and areas where damage may cause engine FOD. These requirements also apply to all access panels. Puncture can be prevented by making the skins thicker than a critical threshold or by using a highly conductive expanded copper mesh layer in the laminate.

The minimum skin thickness requirement for lightning strike varies by location on the aircraft, which is typically divided into zones of lightning severity and probability. A lightning leader initially attaches to an extremity, such as the nose, wingtip, inlet duct openings, or canopy edges. As the aircraft flies through the lightning flash, the flash can be conducted, sweep along the skin, or reattach itself to the skin before it exits out another extremity, such as the tail. The most severe lightning strike conditions occur in Zone 1, which consists of extremities where a lightning leader is most likely to attach or exit and regions where the leader is likely to sweep. Zone 2 consists of regions of lower severity where the lightning leader is likely to continue sweeping or reattach. Zone 3 consists of regions of lowest severity where lightning may continue to sweep across the skin. While program-specific requirements may be higher when imposed by a customer, Lockheed Martin Aeronautics minimum thickness values to preclude lightning protection in these zones are given in Table 2.3-3. Note that for cored sandwich skin configurations, the table specifies the minimum thickness of the top facesheet only—not the total skin thickness. Configuration specific lightning strike testing is highly recommended and can substantiate thinner laminates.

Table 2.3-3—Minimum Composite Laminate Thickness to Preclude Lightning Protection Application

Severity Zone	Description	Sandwich Facesheet (Inches)	Solid Laminate (Inches)
Zone 1	Leader attachment and sweep	0.120 inch	0.241 inch
Zone 2	Leader sweep and return stroke	0.070 inch	0.160 inch
Zone 3	Swept lightning channel	0.050 inch	0.080 inch

2.3.2 Laminate Configuration and Fiber/Ply Placement

The intent of this section is to compile a general listing of rule-of-thumb guidelines for composite laminate design for convenient reference. Further detail on theory and application of many of these guidelines is provided in later sections of this manual.

2.3.2.1 Stacking Sequence

	Guideline	Reason	Remarks
1.	The 0º, ±45º, 90º ply angles are recommended for major load carrying structures.	Minimize matrix and stiffness degradation.	Ease of fabrication; current allowables and analysis methods limit designs to use of these ply angles.
2.	The stacking order of plies should be symmetric about the midplane.	Minimize warping and interlaminar shear.	An exception is when local build-ups or doublers are required, in which case the local doublers should be symmetric within themselves but not necessarily the assembly.
3.	In general, biased oriented plies in monolithic and facesheet laminates should be "balanced," which is achieved by maintaining equal numbers of +45 and –45 layers. Laminates shall be balanced except when directed by Structural Integrity to achieve specific bending/twisting coupling response or one of the following: • Non-structural surface plies. • When loads or preferred mechanical response dictate an unbalanced laminate. • Local areas that are in thickness transition (ply drop-off or buildup) areas.	Unbalanced laminates will experience in-plane shear coupling, whereby shear distortions occur when axial loads are applied.	In the case of an unbalanced laminate, the unbalanced ply or plies should be kept near the centerline.
4.	Composite laminates shall conform to the following ply percentage limits: 8 < %0° Plies < 52 40 < %45° Plies < 84 (40% minimum in bolted joint areas) 8 < %90° Plies < 52 Laminates that conform to the above guideline are known as Fiber Controlled Laminates.	Fiber Controlled Laminates have sufficient fibers in all directions to prevent matrix splitting.	In addition, when more than 60% 0° fibers are used, there is a marked tendency for the laminate to split itself longitudinally under axial load.
5.	If possible, avoid grouping 90º plies. Separate 90º plies by a 45º ply.	Minimize interlaminar shear and normal stresses; minimize multiple transverse fracture.	Minimize grouping of matrix critical plies.

	Guideline	Reason	Remarks
6.	Avoid dropping a 0º ply that is adjacent to a 90º ply.	A 90º ply has little load carrying capability relative to the 0º ply.	Apply this rule judiciously, considering the local stress field.
7.	Stacking sequences should minimize interfaces where the angle between adjacent plies exceeds 45 degrees. The recommended separator between a +45° and a -45° is a 0° or 90° ply.	Minimize matrix microcracking in toughened matrix resins during service.	
8.	Wherever possible, maintain a homogeneous stacking sequence and avoid grouping of similar plies. If plies must be grouped, they should be a maximum of 0.0336" total thickness (4 plies of 0.0084 in./ply material or 6 plies of 0.0053 in./ply).	Strength; Minimize edge splitting; Minimize matrix microcracking in toughened matrix resins during service.	If this guideline is violated, use analysis to show the interlaminar shear allowable is not exceeded.
9.	For parts that are made with bidirectional woven fabric, the orientation angle indicates the warp fiber direction.	Prevent warping of cured laminate.	
10.	For shear-stability-critical plate laminates, place angle plies near the outer surface of laminate.	Improve shear buckling efficiency.	See Section 9.4.
11.	For laminate subject to transverse bending loads place plies with fibers oriented in maximum bending direction near the outer surface of the laminate.	Maximize bending stiffness in order to beam loads to substructure.	See Section 9.4.

2.3.2.2 Ply Terminations

	Guideline	Reason	Remarks
1.	When terminating plies, maintain symmetry; drop between continuous plies in the same direction. Maximum strength is achieved if ply terminations in adjacent plies are a minimum of 0.5 inch apart.	Minimize warping and interlaminar shear; develop ply strength.	
2.	Ply terminations should drop plies from each orientation evenly (i.e. do not drop all the 0º □plies, then all 45º plies, etc.). The last ply termination should be the least stiff ply for the loading (i.e. 45˚ ply for axial load or 0˚ ply for shear load).	Smooth transition in stiffness throughout the transition region.	
3.	Wherever possible, do not terminate plies in fastener patterns.	Maintain joint bearing strength; reduce profiling requirements on substructure.	

	Guideline	Reason	Remarks
4.	Avoid ply terminations near stress concentrations such as cutouts, corners, and joggles.	Avoid compound stress concentration.	
5.	Locate ply terminations such that joggles in adjacent plies are minimized.	Smooth transition in stiffness throughout the transition region.	
6.	Exterior surface plies should be continuous.	Minimize damage to edge of ply.	Cover all ply steps with at least one continuous outer ply.
7.	A nominal taper ratio for ply drop-offs of 20:1 is preferred. A minimum taper ratio of 10:1 may be used where geometric and weight considerations require it. Detail strength analyses or positive test results must support application of ply add/drop rates less than 10:1.	Minimize load introduction into the ply drop-off and resulting interlaminar shear stress.	If possible, ply drop-offs should be symmetric about the midplane with the shortest length ply nearest the exterior faces of the laminate. Nominal ply drop is affected by drawing tolerances as discussed in Section 2.4.3.3
8.	No more than two plies should be dropped at any location. This should be interpreted to mean one ply on each side of centerline of symmetry. Only one ply may be dropped at any location if the ply is equal to or greater than 0.013 in. thick.	Minimize load introduction into the ply drop-off and maintain symmetry.	
9.	A maximum of four ply terminations may be made between any two continuous plies. Good design practice is maximum of two ply terminations. See terminations 2,5,8,8 in Figure 2.3-1.		
10.	Sequence the ply terminations throughout the total thickness in order to maximize the distance between ply terminations in adjacent plies. See terminations 1,4,7 and 3,6 in Figure 2.3-1.		
11.	Figure 2.3-1 shows an example laminate with ply terminations illustrating many of the above guidelines.		

```
              45 ---------------------------------------------------------------------------   NOTES:
             -45 ---------------------------------------------------------------------------   This example is fabric
              45 -------------------------------------6  (Ply termination number)              Odd number of plies
             -45 -------------------- 3                                                        Unbalanced OK for fabric
               0 ---------------------------------------------------------------------------   Does not require 90º plies
              45 --------- 1                                                                   Ply t > 0.013"—only one ply drop per location
             -45 ------------------------------------------7
               0 ------------------------- 4
              45 ---------------------------------------------------------------------------
             -45 -------------- 2
              45 -------------------------------5
             -45 ---------------------------------------------------------------------------
midplane---    0 ------------------------------------------------8  (Last ply terminated is least stiff)
             -45 -------------------------------5
              45 ------------------------------------------------8
             -45 -------------- 2          (Max. of four plys terminated between two continuous plies)
              45 ---------------------------------------------------------------------------
               0 ------------------------- 4
             -45 ------------------------------------------7   (Maximized distance between adjacent ply drops)
              45 --------- 1
               0 ---------------------------------------------------------------------------
             -45 -------------------- 3
              45 -------------------------------------6
             -45 ---------------------------------------------------------------------------
              45 ---------------------------------------------------------------------------
```

Figure 2.3-1—Sample Laminate to Illustrates Ply Termination Guidelines

2.3.2.3 Damage Tolerance Considerations

Guideline	Reason	Remarks
1. Damage tolerance may be augmented by		
a) Using the least strength critical plies at or near the OML surface of the laminate of a strength critical component or, alternatively, using a fiberglass scrim ply if there is a necessity for an outer critical ply, e.g. for increased buckling strength.	Minimize strength degradation in case of minor surface damage.	The least critical ply is generally a +45° fabric ply or ±45º tape plies unless the part is critical for shear buckling.
b) Using bi-directional woven fabric outer surface plies.	Localize extent of damage.	
c) Not grouping any particular ply fiber orientation.	Minimize matrix microcracking in toughened matrix resins.	
d) Adding plies of a ductile material or fiberglass on the back face of a graphite laminate.	Limit damage propagation on back face of laminate surface; reduce handling damage.	The addition of these plies must be coordinated with allowables development testing.

2.3.2.4 Ply Splices

	Guideline	Reason	Remarks
1.	Unidirectional tape shall not be spliced perpendicular to the fiber direction.	Improve load transfer between plies.	Unidirectional materials may be butt spliced parallel to the fiber direction.
2.	Woven fabric material should be lap spliced using at least 0.50 inch overlap. 1.0 inch overlap is recommended.	Improve load transfer between plies.	When warp and fill direction properties are significantly different, the lap splice should be perpendicular to the fill direction and fill directions should align on both sides of the splice.
3.	Butt splicing of fabric plies shall only be done in circumstances where a detailed stress analysis has found that this splice type is acceptable. In cases where analysis determines a part does not meet strength requirements with a butt splice, then an overlap splice shall be used or additional plies added.	Improve load transfer between plies.	A butt splice is typical for unidirectional materials and is always parallel to the fiber direction.
4.	Where multiple layers of woven material are used, stagger splices such that they are at least 2 inch apart on adjacent plies and do not coincide within a 5 ply thickness	Reduce stress concentrations at splices.	
5.	An overlap shall be used at an external ply splice. The overlap shall be at least 0.5 inch and face away from the airstream.	Avoid edge damage and erosion.	
6.	Splices should be located in the least critical areas of the structure		Overlap splices of fabrics can cause significant thickness variations, especially where splices intersect. Care should be taken to locate such areas away from other structure.

2.3.3 Adhesive Bonded Joints

	Guideline	Reason	Remarks
1.	Composite to metal joints		See Section 11.3 for further discussion.

	Guideline	Reason	Remarks
	a) Bonding composites to • Titanium (preferred) • Steel (acceptable) • Aluminum (not recommended)	Ensure material compatibility to minimize corrosion; CTE mismatch.	
	b) Bonded step joints are preferred to scarf joints.	Achieve better fit, more efficient load transfer, higher strength, and design flexibility.	
	c) 90º plies should never be placed adjacent to the bondline unless it is also the primary load direction for the joint.	Maximize lap shear strength.	
	d) If possible, do not end more than two 0º plies, whose maximum thickness is not more than 0.014 inch, on any single step surface.	Smooth load transfer.	
	e) 45º or 90º plies should butt up against the first step of a step joint.	Reduce peak stress loads at end of ply.	
2.	Composite to Composite Joints		See Section 11.3 for further discussion.
	a) In general, cocured joints are preferred to secondarily bonded or cobonded joints in terms of joint strength.		
	b) For secondarily bonded parts, machined scarfs are preferred to layed-up scarfs.	Achieve greater accuracy in ply terminations.	
	c) Use cocured bonded subassemblies whenever possible.	Reduces ply count and assembly hours	
	d) Adherends should have similar coefficients of thermal expansion.	Minimize residual stresses on the adhesive.	
	e) When using unidirectional tape, lay up outer plies in contact with the adhesive at 0° or +45° to the principal load direction.	Balance the membrane stiffness of the adherends	
	f) Maintain a Poisson’s ratio mismatch between bonded or co-cured laminates of less than 0.1.	Excessive property mismatches between bonded elements can result in disbond problems.	Deviation from this guideline should be substantiated through test or analysis.

	Guideline	Reason	Remarks
3.	Avoid geometries that generate interlaminar tension, or peel, stresses in adhesive bonded joints.	The interlaminar direction is the minimum strength direction of composite laminate.	Good design practice is sufficiently long bondlines and tapered adherends. See section 11.3 for further discussion.
4.	Because of adherend elasticity, high adhesive shear stresses are concentrated near the ends of the joint. Ensure the joint has sufficient length such that the adhesive in the majority of the joint stays in the elastic range.	The presence of an elastic trough is important to prevent creep and improve tolerance to bondline defects.	See Section 11.3 for further discussion.
5.	For increased joint efficiency, balance the joint by matching the stiffnesses, measured by modulus x thickness, of the adherends.	This will more evenly distribute load transfer between both ends of the joint.	In unbalanced joints, the majority of the load transfer will take place at the termination of the stiffer adherend and generally overload one end of the joint.
6.	For out-of-plane loading, built-up 2-D and 3-D bonded joints can provide significant advantages in some applications.		A built up bonded joint has section components assembled out of the base laminate plane from 2-D lamina or using 3-dimensional technologies such as weaving. See section 11.3 for further discussion.

2.3.4 Mechanically Fastened Joints

	Guideline	Reason	Remarks
1.	The full bearing capability of a composite material can only be attained using fasteners with high fixity achieved through good clamp-up.	Achieve maximum bearing strength.	Fixity is a function of fastener stiffness, fastener fit, installation forces, torque and rotational resistance of the fastener head and the collar or formed backside.

	Guideline	Reason	Remarks
2.	Use only titanium, Monel, Inconel, A286, PH13-8MO, PH17-4, MP159, or MP35N fasteners with graphite composite laminates	Prevent corrosion of fasteners.	Titanium is preferred; stainless steel is acceptable in low corrosion environment only
3.	Aluminum, cadmium plated steel, or aluminum coated fasteners shall not be used with graphite composite.	Prevent corrosion of fasteners.	
4.	Steel fasteners in contact with graphite should be permanent and installed wet with sealant.	Prevent corrosion of fasteners.	
5.	Use a layer of fiberglass fabric as a galvanic corrosion barrier ply on faying surfaces of graphite composite panels to aluminum.	Prevent corrosion of aluminum.	
6.	Paint both faying surfaces of mechanically attached graphite composite panels to aluminum substructure.	Prevent corrosion of aluminum.	
7.	Fasteners should never be driven or bucked to assemble composites.	Prevent potential damage to composite laminate.	
8.	Layup areas subject to bearing loads should be at least 40 percent 45º plies.	Ensure adequate laminate bearing strength.	Margins computed using IDAT/IBOLT tool take precedence.
9.	Design countersink depth should not exceed 80% of the nominal structural thickness.	Eliminate knife edges in design configuration and avoid stress concentration.	Programs may have additional guidelines; deviations from these guidelines may occur during MRB using margins calculated by IDAT/IBOLT tool.
10.	Fastener edge distance shall be 2.5D minimum, where D is the hole diameter and edge distance is from the hole center to the effective edge of the part. Use 3D minimum edge distance for major-mate joints.	Prevent shear out and net tension failure; obtain pure bearing failure.	Somewhat reduced edge distance may be acceptable where loading is light. See Section 2.4.3.5 for tolerance considerations.
11.	Fastener pitch should be no less than 4D and no greater than 8D, where D is the hole diameter.	Ensure adequate strength and fuel-sealing and prevent local buckling.	Other requirements may dictate smaller maximum or larger minimum pitch within this range.

	Guideline	Reason	Remarks
12.	Use fasteners designed for composite installation for blind attachments to composite substructure or wherever the collar bears against a composite laminate.	Prevent damage to composite laminate by locking collars of fasteners.	These fasteners will improve clamp-up, bearing and pull-through strengths over smaller-footprint fasteners.

2.3.5 Sandwich Structures

	Guideline	Reason	Remarks
1.	Analyze sandwich structure for face sheet strength, panel buckling, deflection, shear crimping, face sheet dimpling, face sheet wrinkling, core shear stress and core crushing.	Check all failure modes	
2.	When possible avoid laminate buildup on the core side of the laminate.	Minimize machining of the core; ensure fit of core to face sheet.	
3.	Sandwich face sheet laminates should be symmetric and balanced about their respective mid-planes.		Guideline excludes taper and transition zones.
4.	Continuous plies should be used at the core-to-laminate interface and OML.		Ply EOP's over the core are permissible but must follow all related guidance.
5.	The laminate facesheets on either side of honeycomb core may be fabric (preferred) or tape. If tape is used in the facesheet then the outermost structural plies and the plies adjacent to the core should be fabric in a 45° orientation.		
6.	For honeycomb sandwich panels with fastener-edge ply doublers, use the following instructions: a) Use ± 45° bi-directional woven fabric, subject to Stress approval. b) Add 3 or fewer doubler plies with steps to the outer skin face sheets. c) Add filler plies butted to the slope of the core.	Prevent shear out and obtain higher bearing allowable.	These practices prevent extra core machining and lower layup cost.

	Guideline	Reason	Remarks
7.	Honeycomb core		
	a) The preferred maximum machined core chamfer is 20º. Larger angles require stabilization	Prevent core collapse.	
	b) For flex core, use a 20º chamfer. This core requires extensive stabilization.	Prevent core collapse.	
	c) Use only non-metallic, titanium, or corrosion resistant aluminum honeycomb core in composite sandwich assemblies.	Prevent core corrosion.	
	d) The choice of honeycomb core density for sandwich structure with bonded or co-cured face sheets should satisfy strength requirements for resisting the curing temperature and pressure—3.1 pounds per cubic foot (pcf) minimum for typical depth structures.	Prevent crushing of core.	Deep structures may require denser core.
	e) Heel damage to the walking surface of sandwich structure is strongly influenced by core density. 3.1 pcf core density is unacceptable. 6.1 pcf core density is preferred.	Avoid damage in walking area	The face sheet on the walking surface should meet minimum thickness guidelines specified in Section 2.3.1.
	f) Do not use honeycomb core cell size greater than 3/16 inch for co-curing sandwich assemblies. 1/8 inch cell size is preferred.	Prevent dimpling of face sheets	
8.	Do not use carbon facesheets with aluminum core greater than 7 lb density without demonstration tests at cold temperatures.	Prevent node disbonds.	
9.	Do not use glass cores greater than 7 lb density without demonstration tests at cold temperatures.	Prevent node disbonds.	
10.	Do not use aluminum and composite cores in the same bonded assembly.	Prevent node disbonds and corrosion.	

2.3.6 Electric Bonding and Lightning Protection

	Guideline	Reason	Remarks
1.	For Electromagnetic Interference (EMI) seals and electric bonding, the shielding device must be the sacrificial and replaceable element.	Provide for easier maintenance and repair.	

	Guideline	Reason	Remarks
2.	Electric bonding and lightning protection shall be in accordance with applicable Engineering and program Electromagnetic Environmental Effects (EEE) design standards.	Each composite part must be electrically bonded to ensure the entire primary structure of the airframe is electrically connected.	Non-structural conductive systems can be used to ensure adequate lightning protection.
3.	The preferred solutions for lightning strike protection are sufficient skin thickness or an Expanded Copper Foil (ECF) ply.	Prevent skin burn-through	Minimum skin thickness requirements for lightning strike are given in Section 2.3.1.

2.3.7 General Guidelines

	Guideline	**Reason**	**Remarks**
1.	Make corner radii as large as possible. Minimum corner radius should be the larger of 0.12” or the thickness of the laminate.	Sharp corners cause high interlaminar stress under bending loads. They also are difficult to fabricate and drape, resulting in bridging and wrinkling of the prepreg and thus weakening of the part.	A larger radius may be required in some cases to meet interlaminar strength requirements or for manufacturing or inspection reasons.
2.	Integral joggle features require detail analysis and risk reduction testing to support design incorporation.	Integral joggle features induce out-of-plane stresses.	Minimum taper ratio should equal or exceed 5:1. Design applications on fracture critical parts should be supported by tests that include damage tolerance and durability.
3.	Limit use of epoxy composites to maximum service temperature of 200º F to 250º F depending on the specific resin system and the temperature supported by hot, wet allowables.	Matrix strength is reduced at temperature.	The maximum operating temperature must be at least 50º F below the wet glass transition temperature of the material.
4.	Limit use of BMI composites to maximum service temperature of 325º F to 450º F depending on the specific resin system and the temperature supported by hot, wet allowables.	Matrix strength is reduced at temperature.	The maximum operating temperature must be at least 50º F below the wet glass transition temperature of the material.

	Guideline	Reason	Remarks
5.	Uni-woven and bi-directional woven fabrics have reduced strength properties and prepreg material of this form costs more than tape prepreg. Fabric should be used only when justified by trade studies except when fabrication costs are less. If justified, woven fabric may be used for ±45º or 0º/90º plies.	Minimize fabrication costs; necessary for complex shapes.	See damage tolerance guidelines in Section 2.3.2.3. For enhanced hand lay-up producibility, specify fabric in lieu of ±45° tape plies.
6.	Eliminate all potential water traps.	Prevent potential galvanic corrosion.	
7.	Use a fiberglass scrim ply wherever laminates are prone to galvanic corrosion through contact with dissimilar materials.	Prevent potential galvanic corrosion.	Because of increased dependence on material and processing control, avoid dissimilar material designs where possible.
8.	Ensure adequate protection between composite panel edges and aluminum skins.	Prevent potential galvanic corrosion.	Fill gaps with sealant.
9.	Access Doors and Panels		
	a) Frequent-access composite doors should be nonstructural, hinged, and secured with quick-opening latches.	Improve accessibility; avoid fastener-removal damage.	Frequent access to be defined by program; could be defined as accessed each 40 flight hours or less.
	b) Limited-access composite doors should be installed with program-specified quick-access panel fastening systems that include grommets or sleeves in the access panel.	Prevent wear and mechanical damage to composite and prevent fastener corrosion.	Limited access to be defined by program; could be defined as accessed between 40 and 400 flight hours.
	c) Semi-permanent composite covers should be structural fit and secured with tension head flush screws without grommets or sleeves.	Decrease cost and weight.	Semi-permanent to be defined by program; could be defined as accessed every 400 flight hours or more.
	d) Sleeves or grommets should be installed in frequent- or limited-access fastener holes.	Prevent fastener head pull-through the laminate under torquing or other tension loading conditions.	The sleeve should be mechanically fastened with captive pins for frequent access doors or covers. In the event that the removable panel is attached to composite understructure, a sleeve should also be used in the understructure.

	Guideline	Reason	Remarks
e)	Minimum 0.25” diameter fasteners should be used to secure all access doors.	Prevent torquing off fastener heads.	
10	Interlaminar shear strength checks must be performed for local pad-ups used to reinforce panel cutouts and load introduction points.	These details redistribute loads locally through interlaminar shear.	
11	Structural designs and loads that induce stresses normal to the laminate should be avoided. Avoid designs where loads subject sharp corners to bending.	These designs create interlaminar shear and tension stresses.	Curved laminates such as channel sections may have high interlaminar tension stresses induced in the corners.

2.4 Accepted Methods of Composite Strength Analysis

The precedence of approved stress analysis methods for new programs is outlined in Section 2.3 of PM-4057, Reference 2-1. This precedence holds for composite structural analysis, with PM-4056 being the first source as it is tailored for use on the program through program-specific and/or customer-generated guidance. Additional references specific to composite structures may also be appropriate and may be cited in accordance with the precedence outlined in PM-4057. Such composite analysis methods references include References 2-2 and 2-3.

2.4.1 Sign Convention Issues

Primary issues of concern regarding sign conventions in stress analysis are defined in Section 2.3.1 of PM-4057, Reference 2-1. Items discussed in that section include Global Aircraft Coordinate Systems, Finite Element Model Coordinate Systems, and General Structural Analysis Sign Conventions. All details discussed in those sections apply to the analysis of composite structures as well.

In addition to the referenced discussion, analysis of composite structures must accurately define and reference lamina and laminate coordinate systems since the analysis must correctly include directional stiffnesses and strengths in order to generate accurate margin of safety calculations. The convention for lamina coordinate system definition is given in Section 3.5.1 of this manual. The convention for laminate coordinate system definition is given in Section 4.3.1 of this manual.

Consideration must also be given to coordinate systems in Finite Element Analysis (FEA) of composite structures. Since material properties are directionally dependent, the global coordinate system cannot be used for material property definition. Local coordinate systems must be defined relative to structural layout and primary load direction in order to properly orient composite material directions with respect to the structure.

2.4.2 Margins of Safety

Margins of safety are defined and discussed in Section 2.5 of PM-4057, Reference 2-1. That discussion addresses calculation of margins of safety, use of interaction curves, presentation of various forms of interaction equations, presentation of interaction curves, and use of combined stresses for margin calculations on compact structure. While the definition of margins of safety and the use of interaction curves is consistent across all structures, margin of safety calculations in composite structure rely on analysis routines and empirical data based on extensive testing.

Consequently, margin calculations for composite structure are performed for specific criteria using tools calibrated to test data. This section defines in brief four basic margins that must be calculated for composite skins: gross acreage unnotched strength, bolted joint strength, damage tolerance, and durability and repair. The discussion in this section is intended to provide a brief overview and not comprehensive theory. More in-depth discussion is provided in later sections of this manual as referenced in the sections below.

Additionally, margins must be calculated for composite details such as honeycomb core, joggles, adhesive bonded joints, composite stiffeners such as hat stiffeners, and runouts at stiffener terminations. Margin calculations for these features are configuration specific and discussed in the relevant sections of this manual.

Buckling and deflection of composite structures are not covered in this section because these analyses frequently do not generate a true margin of safety. As discussed in Section 2.5 of Reference 2-1, the margin of safety indicates the degree to which the structure is able to sustain the anticipated loads of operational use. A true margin of safety is linearly proportional to load magnitude and implies a structural failure at a value of zero[1]. Many structural stability and deflection analyses relate to aircraft performance or other requirements that do not imply structural failure but rather a failure to meet structural design requirements. These requirements include, for example, aerodynamic smoothness and stiffness. Consequently, structural capability for these requirements is typically expressed in terms of a ratio to requirements instead of a margin of safety. Note that some structural stability analyses, such as crippling, do imply structural failure and a margin of safety should be calculated for such failure modes. Buckling criteria for composite structures are given in Section 2.4.6. Stability and deflection of composite structures are discussed in the basic theory provided in Sections 6.4, 7.3, 8.4, and 9.4 of this manual.

2.4.2.1 Gross Acreage Unnotched Strength

Laminate unnotched strength does not typically size laminate gross acreage because durability and damage tolerance requirements dictate greater thickness. While it is generally non-value-added to size the gross-acreage of the laminate to unnotched static allowables in the initial static sizing step only to later resize it to filled hole tension (FHT) and filled hole compression (FHC) allowables per the durability requirement, it is considered important to write a nominal mid-bay unnotched strength margin of safety in order to properly attribute any weight impact of FHT and FHC allowables to the durability requirement and thereby properly direct any weight-mitigation efforts toward durability, rather than static strength, improvements.

Unnotched strength margins are calculated using classical lamination theory to determine point strains and predict failure using composite failure criteria. In addition to standard mechanical loads, this strength margin should consider load contributions from issues such as flutter, bay pressures, and thermal effects. For the maximum strain failure theory, which is discussed in detail in Section 3.5.7.1 of this manual, the ply-based margin of safety (M.S.) is calculated by comparing the mechanical strain ($\varepsilon_{Mechanical} = \varepsilon_{Total} - \varepsilon_{Residual}$) to the allowable strain remaining after residual strains are subtracted ($\varepsilon_{Allowable} - \varepsilon_{Residual}$) as shown in Equation 2.4-1 below. Residual strains in a laminate are those caused by temperature and moisture differential from the stress-free state, or state at cure, as discussed in Section 3.6.2.

$$M.S. = \frac{\varepsilon_{Allowable} - \varepsilon_{Residual}}{\varepsilon_{Total} - \varepsilon_{Residual}} - 1 = \frac{\varepsilon_{Allowable} - \varepsilon_{Residual}}{\varepsilon_{Mechanical}} - 1 \qquad \textbf{Equation 2.4-1}$$

Where
M.S. = margin of safety
$\varepsilon_{Mechanical}$ = mechanical strain
$\varepsilon_{Allowable}$ = allowable strain
$\varepsilon_{Residual}$ = residual strain
ε_{Total} = total strain

[1] A margin of safety of zero should include an appropriate factor of safety, usually 1.5 for ultimate load, as discussed in Section 2.5 of Reference 2-1.

The Margin of Safety reflects how much mechanical strength remains given a residual condition. Details of classical lamination theory, point stress analysis, and laminate failure theories are discussed in detail in Section 4.5 of this manual.

Unnotched fiber strength margins of safety should be calculated in terms of strain at locations operating at high strains or where small radii corners create steep strain gradients not captured in detailed finite element models used to develop loads. For example, chevron notch areas are typically critical due to sharp radii. For chevron features, highly refined breakout finite element models can be used to recover strains at a characteristic dimension, R_c, which is the distance from the edge of a stress singularity to a point at which the unnotched fiber strain allowable can be applied as discussed in Section 4.5.5. Additionally, interlaminar shear strength margins should be calculated at any features that could induce interlaminar shear stresses, such as integral joggles, large corner radii, or ply-drop ratios less than 10:1.

Laminate unnotched strength margins of safety are typically obtained from the IDAT/SQ5 program using finite element loads. SQ5 performs classical laminated plate analysis, including thermal residual stresses, and has the following key features:

- Provides point stress analysis for applied N_x, N_y, N_{xy}, M_x, M_y, M_{xy}.
- Plots laminate stress interaction diagrams.
- Performs interlaminar shear stress analysis.
- Uses secant value of G_{12} for more accurate strength prediction.
- Ignores first transverse ply failure for strength prediction.
- Uses maximum strain failure theory.

Further details about SQ5 can be found in Reference 2-21.

2.4.2.2 Bolted Joint Strength

Details of composite bolted joint strength and analysis methodology are discussed fully in Section 11.2. Only a brief description is provided here to describe the margin of safety requirement. Composite laminates in bolted joints must have margins of safety determined for bearing and bypass strength, interlaminar shear stress, and fastener pull-through.

2.4.2.2.1 Bearing and Bypass Margin

Composite bearing and bypass strength margins of safety for bolted joints are typically calculated using the IDAT/IBOLT program. IBOLT predicts the static strength of a composite laminate in the immediate vicinity of an individual fastener hole by performing a fracture-mechanics-based static strength prediction for a rectangular $[0/\pm45/90]_C$ composite joint element subjected to a combination of biaxial membrane loads, shear loads, and a bolt load acting at a specified angle. IBOLT also treats out-of-plane bending moments. IBOLT calculations are calibrated to composite bolted joint element test data to provide an accurate prediction of joint strength.

The basic strength model used in IBOLT predicts failure of tangential fibers near the fastener hole. A separate check on bearing strength for the no-bypass condition, called the bearing cutoff calculation, has also been included in IBOLT. The primary method of bearing cutoff calculation uses a beam-on-elastic-foundation analysis to provide a bearing stress profile through the thickness of the laminate. A secondary method of bearing cutoff calculation applies a series of factors to account for diameter, countersunk fastener head, e/D, W/D, and %0° plies, and %90° plies. The secondary bearing cutoff analysis method applies to the rigid substructure case, for which the primary method can over-predict the bearing strength.

Other effects treated in IBOLT analysis include a range of joint configuration effects. Joint configurations, single- or double-shear, are modeled by a beam-on-elastic-foundation analysis to account for the thickness and stiffness of the joint members as well as the bolt bending and shear stiffness. Additionally, empirical equations are included to account for fastener head geometry, effects of filled versus open holes, fastener/hole clearance, countersink depth, and fastener torque.

IBOLT generates margins of safety for the applied bearing/bypass loading as well as for the no-bypass, or bearing cutoff, loading condition. For countersunk fasteners, margins are calculated for both the flush, which includes protruding, countersunk condition as well as for the sub-flush countersunk condition. Further details about IBOLT can be found in Reference 2-22.

2.4.2.2.2 Interlaminar Shear Margin

The next margin of safety that must be calculated for composite bolted joints is interlaminar shear stress (ILS). ILS should be checked at all fastener locations near part edges. The reference location for ILS analysis is at the fastener hole center. ILS analysis typically assumes a single fastener row configuration with one free edge and a couple reaction using a triangular load distribution. ILS is calculated using Equation 2.4-2.

$$\tau_{ILS} = \frac{3}{2}\frac{V_{ILS}}{t \cdot w_{effective}}$$ **Equation 2.4-2**

Where
τ_{ILS} = interlaminar shear stress (psi)
V_{ILS} = interlaminar shear load (lbs)
t = thickness of the composite laminate (in)
s = fastener spacing (in)
e = edge distance for heel/toe reaction; max cutoff e = 2·D where D = nominal fastener shank diameter (in)
$w_{effective}$ = min(s, 2·e) is the effective width for the ILS stress calculation (in)

The ILS margin of safety is calculated using Equation 2.4-3

$$M.S._{ILS} = \frac{\tau_{13}}{\tau_{ILS}} - 1$$ **Equation 2.4-3**

Where
τ_{13} = B-basis interlaminar shear stress allowable for the composite laminate. This value is TAU13 in the IDAT material database.

2.4.2.2.3 Fastener Pull-Through Margin

Finally, a margin of safety must be calculated for a fastener pulling through the skin. The fastener pull-through load, or p_{tu}, is the tension load in the fastener that tries to pull the fastener through the skin material. This load is typically induced by panel pressure loading on a skin or cover panel that results in a fastener tension load. This fastener tension usually also includes tension loading that is caused by substructure flange pull-off induced heel/toe effects. The total pull-through load for analysis is the sum of the pull-through loads due to panel pressure loading and due to curvature effects as given by Equation 2.4-4.

$$P = (p_{tu})_{pressure} + (p_{tu})_{curvature}$$ **Equation 2.4-4**

Where
$(p_{tu})_{pressure}$ = pull-through load due to panel pressure loading (lbs)
$(p_{tu})_{curvature}$ = pull-through load due to curvature effects (lbs)

The typical method for fastener pull-through margin of safety calculation in composite laminates is to use the IDAT tool PULLTHRU, which is documented in Reference 2-23. In the PULLTHRU tool, the fastener pull-through shear stress, τ_{PT}, is calculated using the shear area below the fastener head diameter as given in Equation 2.4-5.

$$\tau_{PT} = \frac{P}{\pi \cdot D_{head} \cdot t_{effective}}$$ **Equation 2.4-5**

Where
P = total pull-through load (lbs)
D_{head} = head diameter (in)
$t_{effective}$ = composite laminate thickness (in) below the top of the fastener head. This thickness is less than the total laminate thickness if the fastener is sub-flush.

The fastener pull-through margin of safety is calculated using Equation 2.4-6

$$M.S._{PT} = \frac{TAU13}{\tau_{PT}} - 1$$ **Equation 2.4-6**

Where
τ_{13} = B-basis interlaminar shear stress allowable for the composite laminate. This value is TAU13 in the IDAT material database,which may be accessed when running the PULLTHRU analysis tool.

2.4.2.3 Damage Tolerance

Damage tolerance requirements are defined in Reference 2-16 and specific program tailored structural design criteria such as Reference 2-24. Details of the analysis methodology are discussed fully in Section 12.5 of this manual. Only a brief description is provided here to describe the margin of safety requirement.

The damage tolerance requirement, discussed in Section 2.2.1.2, requires composite structure to sustain design loads in the presence of damage caused by fatigue, corrosion, environment, accidental events, and other sources until such damage is detected, through inspections or malfunctions, and repaired. The once-per-lifetime damage tolerance design load, P_{xx}, is defined by the program certifying authority.

The damage tolerance margin of safety against this requirement is calculated for Fracture Critical composite structure. The check is made on damaged structure loaded at the P_{xx} load level using the IDAT empirically-based composite DADT tool, CDADT. CDADT calculates margins for compression strength after impact (CAI) and tension strength after impact (TAI). Compression after impact is an evaluation of compressive laminate strength remaining after an out-of-plane impact, while tension after impact is an evaluation of tensile strength. Further details about CDADT can be found in Reference 2-25.

Internal loads and moments input to CDADT are typically extracted from the appropriate finite element model and must be averaged based on twice the damage area (circular) predicted by the applicable analysis tool to preclude sizing to peak element force values that often represent a fraction of the actual damage area. Since CAI is not based on a point-failure mode, average strains consistent with gross panel behavior must be used to verify that the damage tolerance requirement is met. Peak strains associated with a gradient or K_t must not be used and the analyst should ensure that loads/moments used in the averaging are not influenced by edge restraints, cutouts, edge notches, or ply ramps.

The CDADT analysis calculates strains and curvatures due to the applied loads, the energy, the area, the DADT margin of safety for CAI, and the DADT margin of safety for TAI. The tool reports an analysis summary listing the critical CAI and TAI values and notes if negative margins were calculated.

2.4.2.4 Durability and Repair

Composite durability and repair requirements are defined in Reference 2-16 and specific program tailored structural design criteria such as Reference 2-24. Details of the analysis methodology are described fully in Section 12.5 of this manual. Only a brief description is provided here to describe the margin of safety requirement.

The durability requirement, discussed in Section 2.2.1.2, requires composite structure to resist mechanical failures or damage modes during normal service operations that could lead to an undue cost-of-ownership burden on the procuring agency. Structural repair requirements are often specified by procuring agencies. Allowance for simple in-service repair of composite structure is a designed-in product attribute. Durability and repair criteria can often be satisfied simultaneously by incorporation of filled hole strain allowables during structural sizing; however, for some laminates durability impact levels may need to be analyzed separately.

Filled hole strain allowables are calculated using IDAT/IBOLT for a 100% by-pass analysis at 150% Design Limit Load (DLL) for typical repair fasteners considering critical environment. Because the IBOLT methodology does not support sandwich construction, analysts should use either empirically derived CAI or facesheet repair strain cut-offs or IBOLT bypass analysis of the sandwich facesheet, depending on program guidance. These calculations are performed across the broad acreage of composite structure, away from edges and stress concentrations.

2.4.3 Drawing Tolerances and Dimensional Considerations

This section provides an overview of drawing tolerance and dimensional considerations relevant to composite structural analysis, outlining some specific issues. It is not intended to be a comprehensive discussion of composite structure design and manufacturing issues. Detailed discussion of design and producibility rules and guidelines are found in the Lockheed Martin Aeronautics Design Manual, Reference 2-26, and program composite design guides, such as Reference 2-20.

A general discussion of drawing tolerances and dimensional considerations relevant to structural analysis is provided in Section 2.6 of Reference 2-1. That discussion covers dimensional considerations due to manufacturing, drawing tolerances, and high variability structure. Issues and guidance provided in that discussion apply to composite structure and the analyst should account for them in composite structural analysis. In addition to that general guidance, some specific additional tolerance and dimensional considerations affect analysis of composite structures and are discussed here. These considerations, which are discussed in the sections that follow, include laminate configuration, laminate thickness, ply terminations and splicing, tooling concept, and the fiber placement fabrication process.

2.4.3.1 Laminate Configuration

Design of a composite structure requires definition of the laminate at all locations. This definition requires specification of the following:

- Material system
- Angles of cross-plied laminae
- Number of laminae at each angle
- Ply stacking sequence

2.4.3.1.1 Material System

First, a composite material system must be specified. Composite material systems and forms in use at Lockheed Martin Aeronautics are discussed in Sections 3.3 and 3.4 of this manual. Structural analysts should coordinate material selection with design, materials, and manufacturing personnel to ensure consideration is given to all requirements on the composite structure. Mechanical, strength, environmental, and physical properties relevant to structural analysis of composite material systems are available through the IDAT/MATUTL tool as discussed in Section 2.4.7.2. Details about MATUTL can be found in Reference 2-27.

2.4.3.1.2 Lamina Orientation

Additionally, each lamina in a laminate must have an angle defined in a reference coordinate system. The standard ply orientations used at LM Aero for most applications are 0°, 45°,90°, and -45°. Limiting ply angle selection to these angles ensures producibility of the laminate and compatibility with structural analysis tools and material strength allowables that have been developed and calibrated at these angles. Design datasets define a rosette as a basis for ply orientation on composite components. A 2-D representation of a rosette is shown in Figure 2.4-1 below. The rosette is an axis system that utilizes the right hand rule as shown in Figure 2.4-2. The rosette is created by defining the rosette origin point and a vector that represents the 0º fiber direction, which is typically aligned with the primary load direction in the component. The rosette uses the 0º fiber direction vector and the laminate skin normal vector to properly orient the rosette spokes. The analyst should ensure that rosettes are defined in engineering datasets in agreement with analysis assumptions and specified correctly, verifying origin, laminate positive normal direction, and 0º fiber direction. Since different analysis tools may have different rosette/sign conventions, the analyst is cautioned to make sure the data are transferred correctly between drawing, analysis, and FEA and analysis codes.

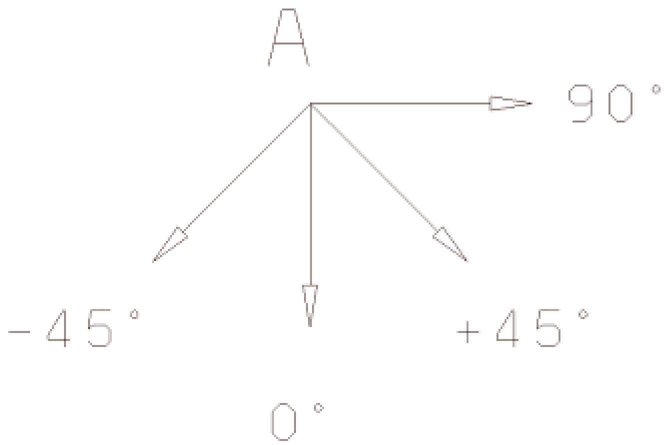


Figure 2.4-1—2-D Representation of Rosette "A"

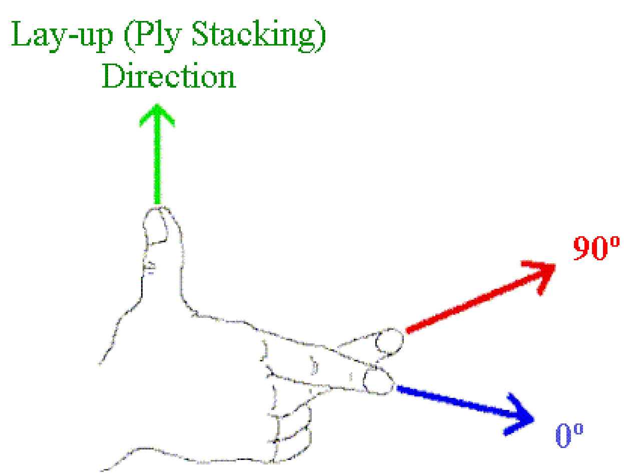


Figure 2.4-2 Right Hand Rule for Rosette Orientation

On engineering datasets, the ply orientation is only held at the location on the part that shows the rosette symbol. Due to the effect of contour on fiber placement, ply orientation may vary away from the rosette location on the part. The recommended tolerance for ply angular orientation to be specified on the field of the engineering drawing is 5°, but this tolerance may vary from program to program.

Additionally, plies in very thin all-fabric laminates exhibit sensitivity to ply surface orientation, or which side faces the tooling, and the cured laminate can exhibit substantial warpage if ply surface orientation is not specified correctly. Consequently, the engineering drawing should specify whether each ply is laid "warp side up" or "fill side up" for very thin fabric laminates or whenever test data indicate a significant sensitivity in properties to ply surface orientation.

2.4.3.1.3 Lamina Count

Next, the number of laminae at each angle must be specified. Ply count at each angle is a discrete number and not subject to tolerance issues. However, these ply counts determine total laminate thickness and tolerance issues with individual plies impact overall laminate thickness dimension and tolerance. Dimensional and tolerance considerations related to laminate thickness are discussed Section 2.4.3.2.

2.4.3.1.4 Stacking Sequence

Finally, the laminae must be ordered into a stacking sequence. The most significant dimensional effects related to stacking sequence occur in laminates that are

- unbalanced, which is where laminae at angles other than 0° and 90° do not occur only in ± pairs
- unsymmetrical, which is where the orientation sequence of plies below the laminate midplane is not a mirror image of the orientation sequence of the plies above the midplane

When the stack is not balanced, symmetric-dimensional issues and, therefore, fabrication, assembly, and structural performance issues can occur as follows:

- Part warpage
- Restraint difficulty during trim, pre-fit and drilling
- Part interchangeability reduction
- Increased potential for cracking vertically through plies
- As the part becomes thicker:
  - Warping reduced
  - Harder to straighten
  - More likely to warp mating structure
  - More likely to promote delamination at fasteners

Refer to Sections 4.3 and 4.4 of this manual for a discussion of the theoretical basis of the behavior of unsymmetric and unbalanced laminated composite plates.

2.4.3.2 Laminate Thickness

For the structural analyst, composite laminate thickness is typically driven by nominal ply thickness and ply count. The analyst and designer specify the laminate configuration as discussed previously in Section 2.4.3.1. Nominal ply thickness is specified in the IDAT/MATUTL material file, which is discussed in Reference 2-27 and used for analyses executed within IDAT composite analysis tools. Most composite stress analyses are executed at nominal thickness, which is sufficient unless a dimension has a tolerance that is an excessive percentage of the mean dimension, as discussed in Section 2.6.2 of Reference 2-1, or the design or material system exhibits high variability, as discussed in Section 2.6.3 of Reference 2-1. Thickness variability in composite laminates comes from either the manufacturing cure process or geometric features that cause fabrication difficulties.

Cured laminate thickness is affected by the following factors:

- Fiber areal weight
- Initial resin content
- Amount of resin removal (bleed or loss) during cure cycle
- Cure cycle pressures
- Type of tooling (matched tooling or bagged surfaces)
- Type of processing

Because of these factors, the specifications controlling the fabrication and each individual material system produce parts with different thickness variations. For detailed discussion of these specifications and variations, the analyst is referred to the Lockheed Martin Aeronautics Engineering Design Manual, Reference 2-26.

In short, laminate thickness tolerance depends on individual cured ply thickness of the material and the cumulative tolerance based on the number of plies in the material. See Table 4.1.3-5 for per ply thickness for some composite materials used by Lockheed Martin Aeronautics. The analyst should consult with program Materials and Processes specialists to determine the proper tolerance limits and the total laminate thickness tolerance for the materials being considered for use and then ensure that thickness values used in analysis follow procedures in Sections 2.6.2 and 2.6.3 of Reference 2-1. The material or process specification provides the nominal per ply thickness and tolerance. Generally, a negative tolerance of 8 to 10% of total nominal laminate thickness is recommended.

Table 4.1.3-5 Thickness of Some Composite Materials Used at Lockheed Martin Aeronautics (For Reference Only)

Material Specification	Material Description	Form	Nominal Thickness (Inch per Ply)	Tolerance or Range (Inch per Ply)
FMS-2023	Carbon/Epoxy	Tape	0.0055	±0.0004
5PTMAT02; LMA-MA002	Carbon/Epoxy	Tape	0.0053	±0.0003
5PTMAT02; LMA-MA002	Carbon/Epoxy	Tape	0.0083	±0.0005
5PTMBT02; LMA-MB002	Carbon/Epoxy	Fabric	0.0083	±0.0007
5PTMAT01; LMA-MA001A	Carbon/BMI	Tape	0.0053	±0.0003
5PTMAT01; LMA-MA001A	Carbon/BMI	Tape	0.0083	±0.0005
5PTMBT01; LMA-MB001	Carbon/BMI	Fabric	0.0083	±0.0007
FMS-1023	Glass/Epoxy	Fabric	0.0095	±0.0002
FMS-1025 Cl I	Glass/Epoxy (Wet Layup)	Fabric	0.009	±0.0002
FMS-1025 Cl II	Glass/Epoxy (Wet Layup)	Fabric	0.0088	-0.0000 +0.0001
FMS-1025 Cl IV	Glass/Epoxy (Wet Layup)	Fabric	0.009	-0.0003 +0.0002
FMS-2003	Boron/Epoxy	Tape	0.005	

The analyst should ensure laminate thickness tolerance studies confirm acceptable ranges for countersinks and shim gaps. The maximum countersink dimensions must not exceed those which produce a knife-edge at the bottom of the countersink when applied to the minimum member thickness. Additionally, there should be no excessive shimming requirement arising from laminate thickness tolerance so the parts will fit properly and time consuming analyses to justify expensive tool changes are avoided. Additional shimming and through-thickness tolerance considerations based on the use of OML tooling are discussed in Section 2.4.3.4.

In addition to manufacturing cure processes, laminate thickness exhibits variability around geometric features with high curvature or ply buildup, such as around ply terminations and splices or cocured features. The analyst should ensure that such features are not located in highly loaded regions or fastener patterns. Sharp corners in composite laminates not only generate high interlaminar stresses but also result in bridging and or wrinkling of fibers, causing voids, thickness variation, resin richness or porosity, and local degradation of laminate stiffness and strength.

To avoid such issues, the analyst should ensure that corner radii are as large as possible, taking into account flange width and hole-to-edge distance constraints. For graphite fibers, 0.25 in. radii or greater are preferred. Recommended minimum corner radius is the larger of the following two values: 0.12 in. or the thickness of the laminate. Plies and core materials should not be dropped in the radius or closer than t, where t is the laminate thickness, from the tangency point of the outer radius. The typical corner radius locational tolerance is ±0.03 in. at the tool surface and the analyst should account for this tolerance in calculations in accordance with procedures in Sections 2.6.2 and 2.6.3 of Reference 2-1. Additional discussion on thickness variation due to ply termination and splicing is found in Section 2.4.3.3 that follows.

2.4.3.3 Ply Termination and Splicing

Ply terminations and splices are another source of tolerance and dimensional considerations within a laminate. Ply terminations, which are identified as end-of-ply (EOP) on the engineering drawing or in the dataset, are the locations within the laminate where plies are added or dropped in order to satisfy laminate strength, stiffness, stability, weight minimization, and any other structural requirements. Ply and core EOP curves have a standard line profile tolerance of 0.12", except where an analysis of mating parts and other factors indicates that a different tolerance is appropriate or if program guidance specifies a different value. The analyst should consider this tolerance and the proximity of the termination to cutouts, fasteners, bends, and any cocured features when ply termination locations are determined in order to preclude unacceptable stress concentrations and ensure smooth load transfer within the laminate. Furthermore, ply terminations create variation in laminate thickness. The analyst should pay close attention to ramp rate in ply terminations, dropping plies as gradually as possible, to ensure smooth load transfer. General ply termination guidelines are given in Section 2.3.2.2.

Additionally, the analyst should consider tolerance and dimensions with ply splices. Material width limitations sometimes necessitate the use of multiple pieces of material to make a complete ply within a laminate. Splices are the interfaces within a ply between two or more pieces of material in order to create a ply of the necessary size. Like ply terminations, ply splices create variation within the laminate in terms of location and thickness. The analyst should consider the proximity of the splice to cutouts, fasteners, bends, and any cocured features when ply splice locations are determined in order to preclude unacceptable stress concentrations and ensure smooth load transfer within the laminate. Splices should be located in the least critical areas of the part and should be mapped to the part in coordination with Manufacturing Engineering, Materials and Processes Engineering, Design Engineering, and Stress personnel. Analysts should ensure engineering drawings and datasets show splice locations when the splice location affects the part minimum margin of safety. Alternatively, drawings could indicate "No Splice" regions. Specific program directives should be consulted to know which convention is being followed. General ply splicing guidelines are given in Section 2.3.2.4.

The analyst should also pay close attention to dimensional variation of the outer moldline (OML) at ply terminations, splices, or laminate joints to ensure that such variations do not expose ply edges to the forward airstream. Ideally, the outermost ply in OML skin laminates should be continuous. Where ply terminations and splices on the exterior surface are necessary and allowed by program policy, they should face away from the airflow, as shown in Figure 2.4-3, in order to prevent erosion and delamination of the ply edge. Such splices should use a 0.5 in. minimum overlap. Likewise, laminate joints should account for thickness differences to preclude exposing laminate free edge to airstream and thereby causing erosion and delamination. Avoid forward facing mismatches in the air stream, as shown in Figure 2.4-4.

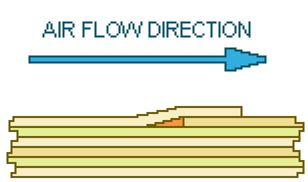


Figure 2.4-3—Outer Mold Line Ply Splice Should Face Away from Airflow

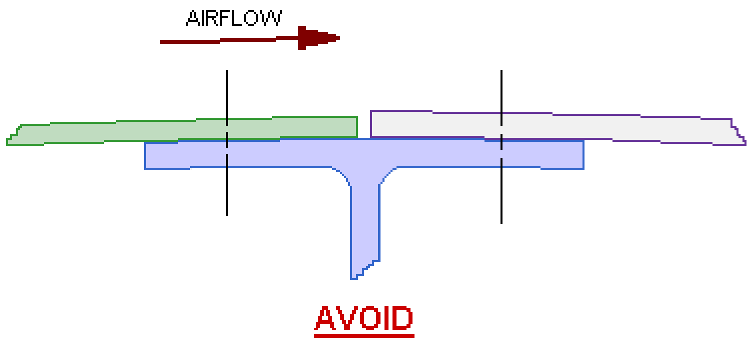


Figure 2.4-4—Laminate Thickness Mismatch Should not Face Forward into Airflow

2.4.3.4 IML vs. OML Tooling

There are many methods and processes for tooling used in the fabrication of composite structure and a full discussion is found in Reference 2-26. One aspect of tooling that impacts structural analysis deals with the side of aircraft skin that is laid against the tooling surface. In general, laminated composite plate structure is manufactured by adding plies to a tooling surface either by hand or through an automated process. Consequently, one side of the composite laminate rests against the dimensionally controlled tooling surface while the other takes the dimensional variation from tolerance stack up. For aircraft skins the tooling surface can be either the inner mold line (IML) or outer mold line (OML). Either of these options has advantages and disadvantages and different implications for structural integrity.

With IML tooling, the inner mold line of the skin acts as the tooling surface and provides primary dimensional control on the inner skin surface as shown in Figure 2.4-5. IML tooling typically assumes maximum material thickness, sets the substructure at the low condition, and allows the OML to float. IML tooling provides producibility advantages that include good fit-up to understructure due to the controlled tool surface and tolerance take out at the OML surface. Disadvantages of IML tooling include the need to machine ply adds and drops into the tool surface, undesired effects on an adequate surface for aerodynamic smoothness on the OML, the need to delay skin tool fabrication until after skin design release, and the need to modify the skin tool for any subsequent skin thickness changes. For structural analysis, IML tooling provides a tighter joint that requires less shimming.

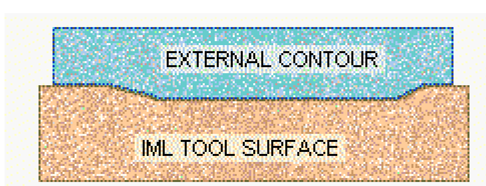


Figure 2.4-5—Inner Mold Line (IML) Tooling

With OML tooling, the outer mold line of the skin acts as the tooling surface and provides primary dimensional control on the outer skin surface as shown in Figure 2.4-6. Advantages of OML tooling include the best external surface smoothness, the ability to make skin tooling as soon as external lines are frozen, and the ability to change skin thickness without requiring tooling changes. As a disadvantage with OML tooling, the understructure mating surface is also the tolerance stack-up surface and typically requires shimming to substructure to ensure an intimate fit between mating surfaces. This shimming is accomplished either through a variety of shimming materials in the form of sheets, straps, or moldable plastic or through sacrificial plies.

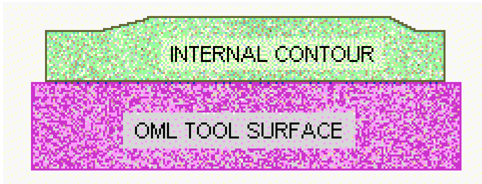


Figure 2.4-6—Outer Mold Line (OML) Tooling

Sacrificial plies are typically woven fabric plies that are added to the laminate in structural joint regions and machined to produce a precisely controlled mating surface on the skins. These machined surfaces are included in the trimmed composite dataset but should be excluded from the structural plies used in analysis because the thickness and number of sacrificial plies remaining after machining varies depending on nominal part thickness, mating part tolerance stackup, and other factors. The analyst should treat sacrificial plies as non-structural shimming in composite joint analyses and should ensure that bolt bending is addressed, if required, in accordance with Section 5.2.3.4 of Reference 2-1.

Another consideration with OML-tooled composite skins is the need to prevent ply drop-offs in sacrificial plies on mating surfaces. Minimum offsets for sacrificial and internal structural ply boundaries from structure are shown in Figure 2.4-7. In order to minimize the size of the composite detail while preventing ply drop-offs in the mating surface due to tolerance variations, the location of the first structural ply-drop, which is distance X in Figure 2.4-7, must be determined by performing a tolerance study accounting for thickness, location, and assembly tolerances of the composite panel and the understructure. The analyst should ensure that tolerance studies, typically performed by Design Engineering, demonstrate that dimension X is such that the first ply drop-off will not fall into the mating surface. Note that a chamfer on the understructure, as shown, may be used to reduce the size of the composite panel.

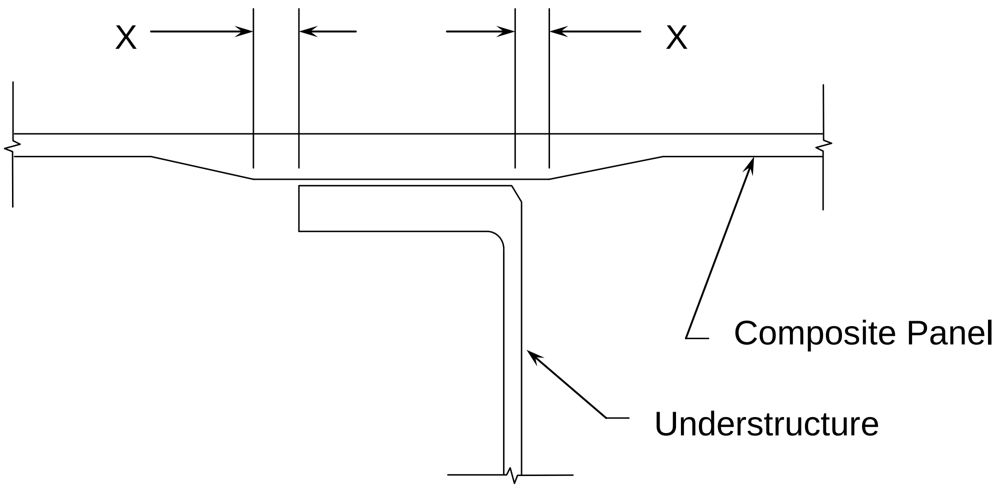


Figure 2.4-7—Composite Panel Minimum Offsets from Structure

2.4.3.5 Fastener Edge Distance

Fastener hole center-to-edge distance guidelines are specified in Section 2.3.4 based on nominal dimensions. Manufacturing tolerance and repairability must also be considered prior to specifying reduced edge distance. Manufacturing tolerance may vary by program and application but could be as large as ±0.03 in. and the analyst must ensure that edge distance based on any negative tolerance condition does not violate minimum edge distance guidelines unless detailed analysis substantiates and program Stress and DaDT management approve the deviation. Such substantiating analysis for reduced edge distances should use minimum tolerance edge dimensions and appropriate reduction factors or reduced allowables for analysis.

The stress and DaDT analysis leads should be consulted when an e/D less than 2.0 is used to ensure that all of the low e/D additional analysis checks have been considered as a part of the static stress and DaDT analysis. Manufacturing and MRB should also be considered when determining the appropriate e/D value for major mate joints, which require larger e/D, and standard repair criteria, which allow for the installation of oversized fasteners. These types of considerations may require additional coordination with design, quality, manufacturing, DaDT and MRB engineering functions.

2.4.3.6 Tooling Thermal Expansion

To avoid mislocating features, tooling for composite structures must provide for the difference in thermal expansion between carbon and most tooling materials other than carbon itself. During a heat-driven cure, the temperature at which the resin solidifies is the gel temperature. At that specific temperature, the part and the expanded tool are the same size. Above the gel temperature, the tool expands more than the partially cured part introducing a thermal strain. As the tool and part cool down from the gel temperature, the tool shrinks more than the part, which also introduces strains. If aluminum is used as a tool, this difference can be enough to cause cracking of the part. While tooling design is developed by the Manufacturing Engineering function, the structural analyst should be aware of tooling design and ensure that thermal expansion differences do not cause manufacturing induced damage.

2.4.3.7 Fiber-Placed Structure

All methods for fabricating composites from 2-dimensional prepreg forms have allowances for overlaps and gaps between ply splices. For hand-laid structures, prepreg plies are typically 12 to 24 inches wide so splices can be effectively spaced in order to avoid coincident overlaps and gaps that can produce regions of material buildup, kinking, or voids that can affect laminate strength. In contrast, relatively narrow prepreg tape used in fiber placement presents more opportunities for material to overlap and gap between tape courses, particularly in regions of geometric complexity.

Basic material elastic and strength properties have been shown to be equivalent between hand-laid and slit (but not towpreg)[1] fiber-placed tape. However, laps and gaps due to fiber placement should be treated as defects (similar to porosity, tool mark-off, etc) and acceptable limits set for each combination of material and manufacturing process. In the case of both fiber placement laps and gaps and co-cured composite facesheets over honeycomb core, certain strength properties may need to be reduced due to these particular manufacturing processes, even when the defects are within acceptable processing limits. Specific knockdown factors, such as those provided in this section, should generally be determined on a program-specific basis.

Assuming that all tows are nominal in width, and spaced perfectly, no overlaps or gaps would occur during the fiber placement process. However, normal slit tape width tolerances of ±0.006 in. result in occasional tow-to-tow gaps and overlaps. Towpreg materials tend to have a larger width tolerance of ±0.010 in. that exacerbates the gap and lap problem. Additional problems have been encountered when the mean width deviates, particularly to the high side, from nominal width. This tends to result in band-to-band overlaps.

Fiber placement process specifications and guidelines, such as References 2-28 and 2-20, allow laps and gaps in fiber-placed structural components up to 0.030 inch and classify laps and gaps greater than 0.100 inch (0.120 inch for gaps resulting from tow cut and add convergence) inches as defects requiring repair. Laps and gaps between 0.030 and 0.100 inch (0.120 inch for convergences) are allowed if the cumulative width of gaps over a 12 inch section perpendicular to the fiber direction is less than the total allowed. The cumulative width of gaps allowed is defined for classes A, B or C in the aforementioned fiber placement process specifications. These gap allowances are reflected in Table 2.4-1, with class A allowing the smallest cumulative width of 0.25 in per 12 inch section. Parts should be zoned into classes A, B, or C in accordance with fiber placement process specifications depending

[1] Slit tape is prepreg cut lengthwise into strips from wider widths of resin impregnated unidirectional fibers. Towpreg is prepreg fabricated from tow which can be converted to woven and braided fabric and used for three-dimensional lay-ups.

on the stress criticality and manufacturing difficulty of the region of the part. Complex geometry generally demands the more generous lap and gap allowance of Class B or C. On the other hand, stress critical structure demands the full material capability of Class A, which places higher demands on manufacturing quality if the geometry is complex. Structural analysts should work closely with Design and Manufacturing Engineering to ensure appropriate zoning to meet structural requirements.

As an example, strength retention factors that have been used on Lockheed Martin Aeronautics programs for each class of gaps and overlaps are shown in Table 2.4-1. The gap/overlap retention factors assume that for every gap there is typically a corresponding overlap, which should be the case if a nominal course-to-course spacing is used. Therefore, the correct amount of material is present in the structure so there is no reduction in stiffness but the increase in fiber waviness can cause a strength reduction. These strength reduction factors apply only to DESIGN and TYPICAL fiber-direction un-notched compression strength, σ_{11C} or ε_{11C} (SIG11(C) or EPS11(C) in IDAT/MATUTL material files), and filled hole compression strength (FHC). These reduction factors should not be applied to IDAT/CDADT analysis, durability/repair analysis or other strength components, such as ILS or tension. When applying this knockdown factor for the FHC allowable, the recommended approach is to maintain a +0.05 (Class B) or +0.10 (Class C) by-pass Margin of Safety in all IBOLT analyses, since the FHC allowable itself is not a user input.

Table 2.4-1— Fiber Placement Strength Retention Factors

Mechanical Property	Class A (0.25 in/12 in.)†	Class B (0.50 in/12 in.)†	Class C (1.00 in/12 in.)†
Modulus	1.00	1.00	1.00
Stress or Strain	1.00	0.95	0.90
†Cumulative inches of lap and gap per 12 inches of placement width.			

Note that any resizing needed to meet this lap/gap requirement should be coordinated with program structural analysis leadership to ensure that all structural criteria are met and the sizing checks have been properly performed to avoid adding unnecessary weight. Viable alternatives in this situation are to rezone the part based on processing trials and/or call out special non-destructive inspection (NDI) procedures in order to explicitly measure laps and gaps per foot of width. The structural analyst should ensure fiber placement classification and quality specifications are in agreement with assumptions used in part analysis.

2.4.3.8 Resin Transfer Molding

Resin transfer molding (RTM) part tolerances depend on tooling. Well-engineered RTM tooling can produce parts with tight tolerances, typically equal to the machine tolerance of the matched metal die tooling.

However, despite potentially good form and fit, RTM parts tend to exhibit regions of resin richness and dryness, particularly at edges and corners, caused by fiber wash, where the resin infusion moves fibers inside the tool. This variation in resin content causes variations in mechanical and strength properties and poor damage tolerance due to susceptibility to matrix cracking. The structural analyst should be aware of these variations because the validity of RTM material allowables, which must be generated specifically for the RTM material system and process, is heavily influenced by the degree of control used in component manufacture. Variability in properties and reduction in allowables typically make RTM an undesirable choice for primary structural components.

RTM cured part fiber volumes are typically lower than in prepreg laminates because of the higher fabric preform bulk factor, which is the ratio of the volume of the unconsolidated fiber preform to the final cured, consolidated volume. A prepreg cloth laminate cured in an autoclave typically achieves a 59% fiber volume. While fiber volumes as high as 58% have been quoted for finished RTM parts, a 55% fiber volume assumption should be used for the design of RTM structure.

For stiffness and strain calculations, assume each RTM ply can take as much load or strain as an equivalent prepreg ply; however, since each ply is laden with additional resin, the load capability per ply is diluted over a larger area.

This results in increased ply thickness, and decreased ply modulus and stress allowables compared to an equivalent prepreg laminate. These property changes are not accounted for in IDAT material files.

Testing of specific RTM applications is needed to accomplish detail structural analysis. However, in the absence of test data, stiffness estimates based on fiber volume ratio may be used for preliminary design of RTM parts. Fiber volume ratios are calculated as shown in Equation 2.4-9 and Equation 2.4-10. These fiber volume ratios are then applied as modification factors to prepreg material properties found in IDAT/MATUTL as shown in Table 2.4-2.

$$\frac{FV_{RTM}}{FV_{preppreg}} = \frac{55}{59} = 0.93$$ **Equation 2.4-7**

Where
FV_{RTM} = typical fiber volume percentage in RTM materials
$FV_{prepreg}$ = typical fiber volume percentage in prepreg materials

$$\frac{FV_{prepreg}}{FV_{RTM}} = \frac{59}{55} = 1.07$$ **Equation 2.4-8**

Where
FV_{RTM} = typical fiber volume percentage in RTM materials
$FV_{prepreg}$ = typical fiber volume percentage in prepreg materials

Table 2.4-2 Monolayer Cloth RTM Properties for Preliminary Lay-up Stacking Analysis

Property	Modification Factor Applied to Prepreg Property
Elastic modulus, E, psi	0.93
Shear modulus, G, psi	0.93
Poisson's Ratio, ν	1.00
Ply thickness, t, in.	1.07

Part of the cost benefit to RTM parts is the ability to stack dry plies of fabric and then use matched tooling to form the shape of the part. The closed tool is then injected with resin. If the part has curved flanges, some of the dry plies may require darting, which results in a decrease in properties unless reinforcement plies are added locally. Reinforcement plies can significantly increase the cost of the RTM part. If no reinforcement is possible, the analyst must account for the reduction in properties, generally through the use of empirically-based reduction factors. Additionally for contoured parts, the fabric can become distorted in-plane which can adversely affect part strength. Testing of the final configuration is recommended.

2.4.4 Statistical Analysis

Statistical methods provide the means to characterize and gain insight into composite materials and structures. Section 2.8 of Reference 2-1 provides an overview of some statistical fundamentals. The discussion in this section assumes the reader is familiar with these fundamentals, which include the following: mean, standard deviation, variance, population, sample, confidence interval, coefficient of variation, probability distribution, cumulative distribution function, normal distribution, tolerance limit, A-basis allowable, B-basis allowable. The reader is directed to Section 2.8 of Reference 2-1 for discussion of these terms.

In addition to these fundamentals, some additional statistical concepts used in development of composite material allowables provide insight into the nature of composites and are briefly discussed here. These concepts include structured and unstructured data, probability distributions used with composite materials, and methods for working with structured data.

The discussion of statistical methods in this section is intended to be conceptual in nature and therefore does not present equations or calculations for the concepts discussed. The details of these methods are considered beyond the scope of this manual. Lockheed Martin Aeronautics develops composite material allowables in accordance with the methods defined in Reference 2-5, MIL-HDBK-17, which presents comprehensive quantitative application and examples of the methods required. MIL-HDBK-17 or its successor, CMH-17, is subject to revision based on state-of-the-art best practices and is considered the source reference for the composite material allowables development process.

Lockheed Martin Aeronautics subject matter experts should be consulted prior to beginning a composite material allowables development program. However, when a program needs to develop allowables from point design testing of composite structure, a simplified approach can be used. Section 2.4.4.4 presents such an approach. Section 18.4 discusses the building block testing approach to determining and verifying material allowables and design values for composite structures.

2.4.4.1 Data Variability and Structure

Data variability and structure plays an important role in statistical analysis of composite material data due to the additive manufacturing and batch cure processes used and the relatively inhomogeneous composition of the cured material. Variability in measured data comes from fixed effects, random effects, random error, and material variability. A *fixed effect* is a systematic shift in a measured quantity, such as tensile strength, due to a particular level change of a treatment or condition, such as test temperature or a machine setting. A *random effect* is a shift in a measured quantity due to a particular level change of an external, usually uncontrollable, factor, such as batch production leading to batch-to-batch differences. *Random error* is that part of the data variation that is due to unknown or uncontrolled external factors and that affects each observation independently and unpredictably. Random error is distinguished from a random effect in that random error is both uncontrolled and unknown, whereas a random effect is uncontrolled but known. *Material variability* is a source of variability due to the spatial and consistency variations of the material itself and due to variations in its processing, such as inherent microstructure, defect population, cross-link density, etc. Components of material variability can be any combination of fixed effects, random effects, and random error.

As a result of these fixed and random sources, composite materials typically exhibit considerable variability in many properties from batch to batch. Consequently, statistical analysis of composite material data must account for batch to batch variability, either by demonstrating it can be ignored or by using a method that addresses it. The concepts of structured and unstructured data are useful in accounting for batch to batch variability and are discussed below.

Data for which natural groupings exist, or for which responses of interest could vary systematically with respect to known factors, are *structured data*. For example measurements made from each of several batches could be grouped according to batch, and measurements made at various known temperatures could be modeled using linear regression; hence both can be regarded as structured data. Structured data that exhibit variation between natural groupings must be modeled using linear statistical models described in Section 2.4.4.3. While these techniques can be used to account for variation between natural groupings, it is often desirable to be able to show that a natural grouping of data, such as by batch, has no significant effect because unstructured data are easier to analyze.

Data are considered *unstructured* if all relevant information is contained in the response measurements themselves, either because of knowledge about the data or because known effects are able to be ignored. For example, data measurements that have been grouped by batch, or panel where possible, and demonstrated to have negligible batch-to-batch variability may be considered unstructured. These data groupings are considered compatible and may be treated as part of the same population. Thus, a structured data set, with a natural grouping identified, can become an unstructured data set by showing that the natural grouping has no significant effect. Data from different batches must be shown to be compatible before they can be assumed to come from the same population. The k-sample Anderson-Darling test, which is discussed in detail in Section 8.3.2 of Reference 2-4, examines if the differences among groups of data are negligible and is useful for determining whether the data should be treated as structured or unstructured. Unstructured data are modeled using a Weibull, normal, or lognormal distribution as discussed in Section 2.4.4.2 below. If none of these distributions are acceptable, nonparametric basis values are determined.

2.4.4.2 Probability Distributions

A probability distribution is a formula that gives the probability that a value will fall within prescribed limits and is used to describe populations of unstructured engineering data so that predictions can be made based on this description. Of particular interest in this manual is the application of probability distributions to fit unstructured engineering test data and determine basis values for composite material system allowables used in structural analysis.

The suitability of a particular probability distribution to describe a population based on a sample of data is determined using a goodness-of-fit test as discussed in Section 8.3.4 of Reference 2-5. This test generates an observed significance level that assesses whether a sample could be from a specified distribution and is used to determine which distribution should be used.

Composite material data are typically represented using a Weibull, normal, or lognormal distribution in determination of basis values for allowables. As discussed in Section 2.8.1 of Reference 2-1, the normal distribution is a good fit for many datasets used in engineering analysis. A normal distribution is defined by two parameters: the mean and standard deviation. Many test statistics are fit well by the normal distribution or some form that can be derived from the normal distribution. However, as discussed in Section 8.3.4 of Reference 2-5, the Weibull distribution is the preferred distribution for the strength distribution of brittle materials such as composite fibers. A Weibull distribution is also defined by two parameters: a shape factor and a scale factor. With these parameters the Weibull distribution can represent skew in the distribution and shape subtleties in the tail region of the distribution. The last distribution used with composite materials is the lognormal distribution, which is related to the normal distribution. If something is lognormally distributed, then its logarithm is normally distributed. In a lognormal distribution , the mean and standard deviation and other operations are calculated and performed using the logarithm of the data rather than the original observations. The computed basis value must then be transformed back to the original units by applying the inverse of the log transformation which was used.

As previously mentioned, the Weibull distribution is preferred for composite material data. If the Weibull model cannot be shown to adequately fit composite material data, then the normal and lognormal tests are performed in succession. If none of these three population models can be demonstrated to adequately fit the data, then nonparametric procedures must be used to compute basis values. Nonparametric statistical methods handle data from samples where nothing is known about the parameters of the variable of interest in the population. In such samples, the data is of low quality, which can arise from small samples or poor fit to distributions. Nonparametric methods do not rely on the estimation of parameters such as the mean or standard deviation.

2.4.4.3 Methods for Structured Data

Composite materials typically exhibit considerable variability in many properties from batch to batch. Because of this variability, one should not indiscriminately pool data across batches and apply the statistical methods for unstructured data. Pooling batches involves the implicit assumption that between-batch variability is negligible, which assumption can produce unconservative basis values in the event it proves false.

Consequently, basis values must account for the sources of variation among natural groupings of data when these sources are shown to be significant by the tests described in Section 2.4.4.1. Details of methods for basis value calculation with structured composite material datasets are given in Section 8.3.5 of Reference 2-5. Two tools used in these calculations are regression and analysis of variances.

Regression methods analyze the relationship between several independent or predictor variables, which are the natural groupings in the data, and a dependent or criterion variable. The objective of a regression analysis for material basis properties is to obtain basis values for a particular response (*e.g.*, tensile strength) as functions of fixed factors (*e.g.*, temperature, lay-up, and humidity). For the simplest case of one independent and one dependent variable, which can be visualized on a two-dimensional scatter plot, a regression line is fit through the data that quantifies the contribution of the independent variable and predicts the response of the dependent variable. This technique is extended to more variables as a plane in an n-dimensional space, where n is the number of independent variables. A wide variety of regression models exist that can be applied to structured datasets.

Analysis of variance (ANOVA) is used to calculate basis values in the simplest case of structured data where the only grouping is by a random effect, such as batches or panels, when there is significant batch-to-batch variability. In the context of composite material allowables, ANOVA is a regression model where the variable levels indicate which batch is associated with each observation. Before basis values are calculated, a diagnostic test for equality of variances should be applied to ensure compatibility of the batches in the ANOVA.

2.4.4.4 Material Allowables Development

Lockheed Martin Aeronautics develops composite material design allowables in accordance with the methods defined in Reference 2-5, MIL-HDBK-17-1, and discussed conceptually in Sections 2.4.4.1 to 2.4.4.3. Testing and statistical calculation of design values must be performed for each composite material system. A full program to develop B-basis design allowables for a composite material system consists of dozens of test series using specific laminate configurations, multiple temperature and moisture conditions, multiple batches, and large sample counts. Execution of such a program requires substantial expertise and Lockheed Martin Aeronautics subject matter experts should be consulted prior to beginning a composite material allowables development program. At Lockheed Martin Aeronautics, composite material system design values are found in the IDAT/MATUTL software tool. Structural analysts should consult the program structures lead engineer for specific guidance on source data for composite material system allowables.

A program sometimes needs to develop configuration specific design allowables from point design testing of composite structure. In such cases, point design testing is conducted on a specific design with specific materials, geometry, and loading, usually with a limited number of test specimens and with limited applicability. In this situation, a smaller number of samples are available but a B-Basis property or allowable is still desired and the normal distribution equations may be used to make the B-basis design allowable calculation. Such an approach should use the methods outlined in Section 2.8 of Reference 2-1.

The example problem from Section 2.8.1 of Reference 2-1 is repeated here for illustration and reference. For a design basis calculation, it is desirable to ensure that some percent of the population does not fall below the calculated allowable. This type of calculation uses a one-sided tolerance interval. Table 2.4-2 provides one-sided tolerance limit factors for varying number of sample sizes to calculate a B-Basis allowable that has a 95% confidence factor. Figure 2.4-8 shows an estimated one-sided normal distribution. Note that to the right of the mean value the curve extends continuously, but to the left it is terminated at a lower bound value.

Table 2.4-3 One-Sided Tolerance Limit Factors for Normal Distributions, 95% Confidence Factor[1]

Sample Size	$K_{one-side}$, B-Basis	Sample Size	$K_{one-side}$, B-Basis
		21	1.906
2	20.581	22	1.887
3	6.157	23	1.870
4	4.163	24	1.854
5	3.408	25	1.839
6	3.007	26	1.825
7	2.756	27	1.812
8	2.583	28	1.800
9	2.454	29	1.789
10	2.355	30	1.778
11	2.276	31	1.768
12	2.211	32	1.758
13	2.156	33	1.749
14	2.109	34	1.741
15	2.069	35	1.733
16	2.034	36	1.725
17	2.002	37	1.718
18	1.974	38	1.711
19	1.949	39	1.704
20	1.927	40	1.698

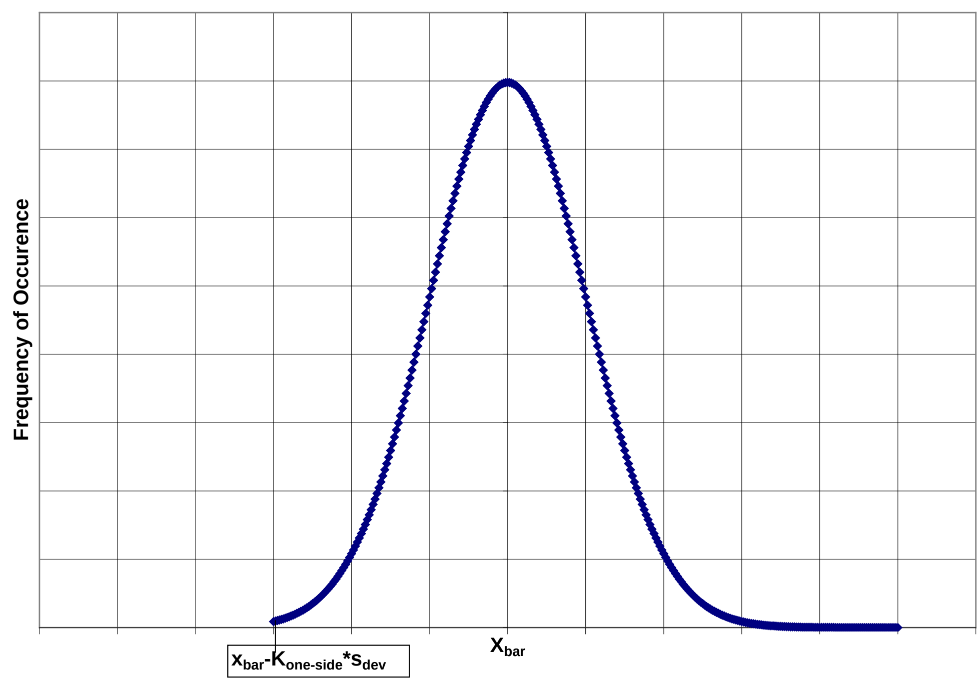


Figure 2.4-8 One-Sided or Single Tail Normal Distribution

[1] Additional values of $K_{one-sided}$ for larger sample sizes can be obtained from Table 8.5.10 in Reference 2-4.

To calculate the allowable using the one-sided tolerance interval, first calculate the sample mean, x_{bar}, from

$$x_{bar} = (x_1 + x_2 + x_3 + \ldots x_N)/N \quad \textbf{Equation 2.4-9}$$

where
x_i is the value of the observation
N is the total number of observations in the sample

Next, calculate the sample standard deviation, s_{dev}, from

$$s_{dev} = [(\Sigma_{i=1}^{N}(x_i\text{-}x_{bar})^2) / (N\text{-}1)\ ]^{0.5} \quad \textbf{Equation 2.4-10}$$

where
N is the number of observations in the sample

Finally, enter Table 2.4-2 with the number of samples and read the value of the one-sided tolerance limit, $K_{one\text{-}side}$ and calculate the B-basis allowable from

$$F_{B\text{-}Basis} = x_{bar} - K_{one\text{-}side} \cdot s_{dev} \quad \textbf{Equation 2.4-11}$$

Where
$F_{B\text{-}Basis}$ is the B-Basis allowable (psi), generically shown as stress but could be load (lbs) or strain (in/in)
$K_{one\text{-}side}$ is the one-sided tolerance limit from Table 2.4-2
x_{bar} is the mean of the test sample from Equation 2.4-7
s_{dev} is the standard deviation of the test sample from Equation 2.4-8

From Table 2.4-2 it is obvious that the smaller the number of samples, the larger the multiplying factor on the standard deviation of the sample and thus the larger the reduction to obtain a B-Basis allowable.

2.4.4.4.1 Example Problem – 10 Observations

Testing has provided 10 observations of failure load (lbs) as follows: 100, 92, 90, 102, 105, 98, 100, 99, 105, 94 Calculate a B-Basis allowable assuming a normal distribution.	
Calculate the mean value of the data population, x_{bar}	(100+92+90+102+105+98+100+99+105+94) / (10) = 98.5
Calculate the standard deviation of the data population	$s_{dev} = [(\Sigma_{i=1}^{N}(x_i\text{-}x_{bar})^2) / (N\text{-}1)\ ]^{0.5} = \{[(100\text{-}98.5)^2+(92\text{-}98.5)^2+(90\text{-}98.5)^2+(102\text{-}98.5)^2+(105\text{-}98.5)^2+(98\text{-}98.5)^2+(100\text{-}98.5)^2+(99\text{-}98.5)^2+(105\text{-}98.5)^2+(94\text{-}98.5)^2]/(10\text{-}1)\}^{0.5} = (236.5/9)^{0.5} = 5.126$
From Table 2.4-2	$K_{one\text{-}side} = 2.355$
$F_{B\text{-}Basis} = x_{bar} - K_{one\text{-}side} \cdot s_{dev}$	$F_{B\text{-}Basis} = 98.5\text{-}2.355 \cdot 5.126 = 86.43$ lbs
The B-Basis allowable with a 95% confidence level for this set of data is 86.43 lbs.	

2.4.4.4.2 Example Problems – 3 Observations

Testing has provided 3 observations of failure load (lbs) with the same mean and standard deviation as Example 2.4.4.4.1: $x_{bar} = 98.5$ and $s_{dev} = 5.126$ Calculate a B-Basis allowable assuming a normal distribution.	
From Table 2.4-2	$K_{one\text{-}side} = 6.155$
$F_{B\text{-}Basis} = x_{bar} - K_{one\text{-}side} \cdot s_{dev}$	$F_{B\text{-}Basis} = 98.5\text{-}6.155 \cdot 5.126 = 66.95$ lbs
The B-Basis allowable with a 95% confidence level for this set of data is 66.95 lbs.	

2.4.4.4.3 Example Problem - 30 Observations

Testing has provided 30 observations of failure load (lbs) with the same mean and standard deviation as Example 2.4.4.4.1: x_{bar} = 98.5 and s_{dev} = 5.126 Calculate a B-Basis allowable assuming a normal distribution.	
From Table 2.4-2	$K_{one\text{-}side} = 1.777$
$F_{B\text{-}Basis} = x_{bar} – K_{one\text{-}side} \cdot s_{dev}$	$F_{B\text{-}Basis} = 98.5\text{-}1.777 \cdot 5.126 = 89.39$ lbs
The B-Basis allowable with a 95% confidence level for this set of data is 89.39 lbs.	

From a comparison of these three examples it can be seen why it is desirable to have a large number of data points. For the same mean and standard deviation, the resulting allowables can vary from 66.95 lbs for 3 observations to 86.43 lbs for 5 observations to 89.39 for 30 observations. More test coupons can result in lighter, more efficient designs. Additionally, the smaller the number of tests, the larger the impact of any single reading is on the standard deviation, leading to a much larger effect of a single low reading on the calculated allowable.

2.4.5 Failure Criteria

Effective failure criteria are essential to predicting the unnotched capability of a laminate in a complex loading environment. The anisotropic and heterogeneous nature of composite materials generates failure modes very different from homogeneous isotropic materials. In the same manner that stiffness varies with direction, the ultimate load a composite lamina can carry also depends on fiber orientation. Laminate failure criteria are based on lamina characteristics, including the orientation and the stress state relative to the strength in each lamina. Lamina failure criteria determine strength and mode of failure of a unidirectional composite or lamina in state of plane stress.

The standard failure criterion applied to laminated composites at Lockheed Martin Aeronautics is the initial or *first ply failure* criterion. It simply states that the laminate has failed when the first ply in the laminate has failed. This criterion is easy to apply, conservative, and is the standard basis for static margins of safety. First ply failure has been used in tool calibration with test data at Lockheed Martin Aeronautics and is used in composite laminate analysis tools in the IDAT toolset.

In many instances first ply failure is not an indication of catastrophic laminate failure, as it may be possible to redistribute the load among the remaining plies without total failure. In a *sequential ply failure* model, failure of a single lamina does not necessarily mean failure of the entire laminate but does cause a change in stiffness. In this model, after a load level is determined that causes first ply failure, the stiffness of the failed ply is reduced or eliminated, the laminate stiffness is recalculated, the load is redistributed among the remaining plies, and ply margins of safety are again calculated. If another ply has a negative margin, its stiffness is likewise reduced and margins again recalculated. This stiffness reduction continues until there are no more negative margins or all plies have failed. The load is increased on the reduced stiffness laminate until the next ply fails and the stiffness reduction is repeated. This continues until all plies in the laminate have failed. The sequential ply failure model is not typically used on Lockheed Martin Aeronautics programs due to the added complexity, lack of tool calibration, and potential for unconservatism in the underlying assumptions.

Laminae, and therefore laminates, have two basic modes of failure. The first is fiber dominated and the second is matrix dominated. *Fiber dominated* failure occurs when fibers break and no longer carry load. Fiber breakage is primarily driven by fiber direction stress or strain in a lamina. A lamina fails catastrophically when the longitudinal tensile strength is exceeded. Using a first ply failure criteria, catastrophic laminate failure is predicted whenever the longitudinal tensile strength of any ply in a laminate is exceeded.

Matrix dominated failure occurs when the matrix fails and fibers are no longer bound together, either within a ply or between plies, thus allowing them to separate. Since the matrix is the binding agent in the composite and is much weaker than the fiber, matrix failures are more prevalent than fiber failures. Matrix failure can occur in many forms and arise from loading in matrix directions, either in-plane transverse to the fiber or out-of-plane interlaminar shear or tension. *Intralaminar* matrix cracks, also referred to as transply cracks or ply splits, run parallel to the fiber direction in-plane and span the thickness of a ply or group of plies stacked with the same orientation. In a tape lamina loaded perpendicular to the fibers, the ply will not carry load after matrix failure. Similarly, a tape lamina loaded in shear will not carry load after matrix failure. However, a tape or fabric lamina can still carry load in the fiber direction with intralaminar matrix cracks present. Consequently, a laminate can carry load with intralaminar matrix cracks if the ply stacking guidelines of Section 2.3.2 have been followed to ensure sufficient plies exist in the loading direction. *Interlaminar* matrix failure, often referred to as delamination, can form near free edges or at intersections between intralaminar cracks. Delaminations form due to excessive interlaminar normal and shear stresses. The accumulation of intralaminar and interlaminar matrix failures depends strongly on laminate stacking sequence.

Numerous theories have been proposed to determine composite material failure, many of which are discussed in standard texts and references such as References 2-7 and 2-8. It is difficult to determine a single strength theory that properly accounts for all of the failure modes found in composite materials and verification of composite failure theories is complicated due to scatter in measured strengths and inconsistencies in techniques. Reference 2-29 compares the suitability of several criteria for predicting failure of various laminates under various loading to provide some insight into composite failure theories. The majority of proposed failure theories use ply level stresses or strains to predict first ply failure. These theories require strength properties in material principal directions: tensile and compressive strengths, or ultimate strain, in fiber and transverse direction and shear strength, or ultimate shear strain. Composite failure theories predict first ply failure and most of them can be used to predict subsequent ply failures using the sequential ply failure model discussed previously.

At Lockheed Martin Aeronautics the maximum strain failure criterion is used for unnotched composite laminate failure prediction. This criterion is applied at the lamina level to determine first ply failure and is discussed in Section 3.5.7 of this manual. Composite material failure is also discussed in several other sections as listed in Table 2.4-4. Under static loading conditions, composites are particularly notch-sensitive as a function of lay-up and, more specifically, stacking sequence. Composite material notched failure criteria are discussed in Sections 4.5.7 and 11.2.7.

Table 2.4-4 Composite Material Failure Topics in PM 4056

Failure Topic	PM 4056 Section
Lamina	3.5.7
Laminate	4.5.7
Laminated Composite Beams and Columns	6.7
Torsion of laminated sections	7.5
Laminated composite rings, frames, and arches	8.5
Laminated plates and shells	9.6
Sandwich	10.6
Mechanically Fastened Joint failure	11.2.7
Bonded Joint failure	11.3.6
Cyclic loading effects	12.6

2.4.6 Buckling Criteria

Most composite aerospace structures are built up from thin plates, which have thicknesses very small compared to their length and width. These built-up structures typically include stiffeners with cross sections that are small compared to their lengths. When loaded in compression, shear, or combined loading that includes compression and shear components, such structures become subject to instability or collapse. Modes of instability can be

- global, where the overall structure of plates and stiffeners collapse together,
- plate, where a plate buckles locally between stiffeners,
- crippling, where a stiffener cross-section fails, or
- a complex interaction among local buckling and overall deformation of plates and stiffeners.

This section addresses items to consider when conducting buckling analyses, considerations for postbuckled composite structure, and allowable buckling loads. Note that crippling is an ultimate collapse failure that has already taken account of postbuckled behavior.

2.4.6.1 Buckling Considerations

Composite skins, webs, and covers must be checked for stability failure in accordance with the criteria stated in section 2.4.6.3. The items to consider when conducting buckling analysis include:

- Loads
  - Allowable buckling load level—see Section 2.4.6.3
  - Pressure loading effects
  - Applied internal loads to analyze for buckling
  - Methods to derive buckling applied loads from internal loads (averaging methods, etc.)
  - Combined loading effects
- Geometry:
  - Panel thickness
  - Panel size and shape
  - Cutouts and pad-ups
  - Edge fixity
  - Panel stiffening (including minimum stiffener sizing for buckling panel breakers)
- Material Properties:
  - Buckling cut-off stress values
  - Elevated temperature material modulus
  - Laminate stacking sequence
- Post-buckling—see Sections 2.4.6.2 and 9.4
- Analysis Theory and Methodology—see Sections 6.4, 7.3, 9.4, and 10.5

2.4.6.2 Postbuckled Structure

A conservative approach in structural design is to not allow buckling below design ultimate load (DUL) in order to preclude stability or collapse related failures of the structure. However, buckling of a structural component does not necessarily imply structural failure because individual components can buckle and still carry load. In many structures, weight efficiency calls for allowing certain skins and shear webs to buckle.

Many studies (*e.g.*, References 2-30 through 2-33) have demonstrated the capability of composite plates and stiffened composite structures to function in a postbuckled state. However, the extent of this postbuckling capability depends on laminate and structural configuration and is a function of material system, ply angle percentages, stacking sequence, width-to-thickness ratio, and cutout size. Additionally, since postbuckling induces out-of-plane stresses, repeated loading into the postbuckling range can cause delaminations to grow if the out-of-plane stress magnitude is sufficiently high. Furthermore, laminates in a postbuckled state exhibit sensitivity to impact damage location due to variation in postbuckling membrane stresses in different locations in the laminate.

While postbuckled structure can be weight efficient, it places additional requirements on structural analysis. Where buckling is allowed, the analyst must ensure flange and stiffener structure is sufficient for carrying the extra compression and bending loads that are present due to the post-buckled state of the web. Flange and stiffener structure must be checked through a separate analysis for the combined effects of the loads that are already carried by the structure plus the additional loading due to the postbuckled condition of all webs adjacent to the flange or stiffener.

Basic theory and analysis methodology for postbuckling of stiffened plates and shells is given in Section 9.4.4.7 of this manual. In performing postbuckled analyses of composite structure, the analyst should consider and include all effects caused by structure that is allowed to buckle. These include, but are not limited to:

- Redistribution of load—consult with Air Vehicle Finite Element Team
- Precipitation of early general instability collapse
- Redistribution of stiffness effect on global flutter—consult with Air Vehicle Flutter Team
- Redistribution of stiffness effect on local (panel) flutter
- Acoustic effects
- DaDT considerations
- Aerodynamic smoothness considerations—see program external features design standard
- Fuel sealing considerations—see program fuel seal design guide

2.4.6.3 Allowable Buckling Loads

Table 2.4-5 provides recommended buckling criteria that have been successfully applied to composite structures on programs at Lockheed Martin Aeronautics. Subject to the limitations in this section, structural panels and webs may be allowed to buckle where affordable weight savings can be demonstrated. The criteria presented are the minimum limits allowed. In general:

- Global stability is required
- No oil canning or snap-through is permitted (refer to Section 9.4 for description and discussion of these phenomena)

Programs that plan to allow buckling of composite structure at lower load levels than Table 2.4-5 should conduct comprehensive testing of relevant structural configurations in order to quantify buckling load limits and constraints. Additionally, programs should coordinate with the certifying authority to ensure concurrence with defined buckling criteria.

Table 2.4-5 Composite Structural Stability Criteria

Type of Structure[1]		Buckling Allowed at Percent Limit Load[10]
Mold Line Skins	Wing, vertical and horizontal tails	120[2]
	Fuselage—solid laminate	37.5[2]
	Fuselage—Syncore stiffened	150
	Control surface	150
Non-Moldline Skins (e.g. nacelle, duct)		150
Periphery Webs[4,5] (incl. interfaces)	With holes	150
	Without holes	37.5[6]
End Bay Webs (incl. Spar, Keel Beam, or Longeron)		150
Non-Periphery Webs[4,5]	With holes	150
	Without holes	37.5
	With reinforced holes[7]	37.5
	With attached subsystem fitting	150
	Fuel Boundary[3]	37.5[3]
Access Panels		150
Core Stiffened Panel (incl. honeycomb & Syncore)		150
Panel Thickness > 0.2 in.[8]		150
Small Radius of Curvature ($r \leq 30$, $e > t$)[9]		120

1 Where more than one classification applies use the most conservative criterion
2 Buckled panel deflection and waviness must comply with program external features design standard
3 A Fuel boundary is defined as a panel or web adjacent to fuel sealant or sealing grooves, see Figure 4.3.2.4-1. These criteria cover a structural basis only. Fuel boundaries shall comply with program fuel seal design guide.
4 See Figure 4.3.2.4-2 for definition
5 Webs adjacent to boundary flanges that rely on in-plane web support for curvature effects should not buckle, see Figure 4.3.2.4-3
6 Periphery webs without holes may buckle only when web stability is not needed for OML flange strength, stability, or stiffness requirements
7 Reinforced hole has local pad up and no system attachments
8 Thickness limit based on upper bound of legacy program experience
9 No compression buckling (loads normal to curvature, see Figure 4.3.2.4-4) with $r \leq 30$ in. or where panel eccentricity exceeds panel thickness, t, based on fatigue experience
10 Buckling under specific load conditions may be considered on an exception basis. These conditions may include: crash, jacking, spin, taxi, emergency arrestment, launch, tie down, and ground handling. Additionally, buckling under any ground handling conditions must not cause functional impairment.

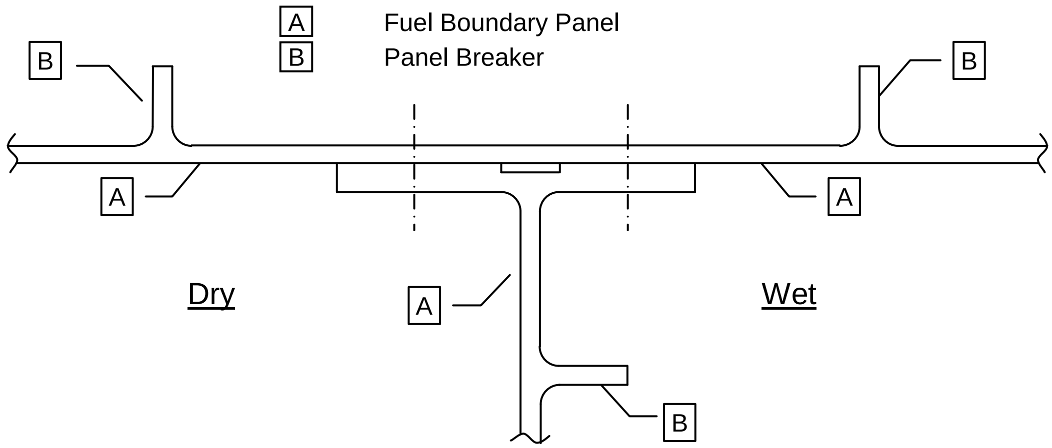


Figure 2.4-9 Fuel Boundary Definition

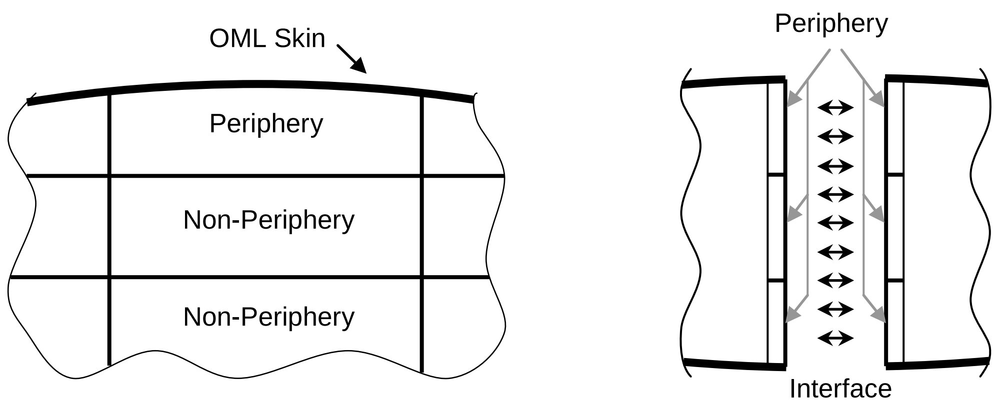


Figure 2.4-10 Definition of Periphery and Non-Periphery Webs

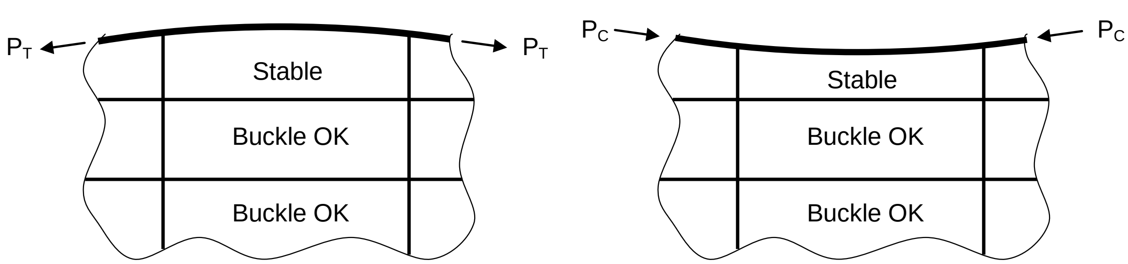


Figure 2.4-11 Definition of Webs Adjacent to Curved Boundary Flanges

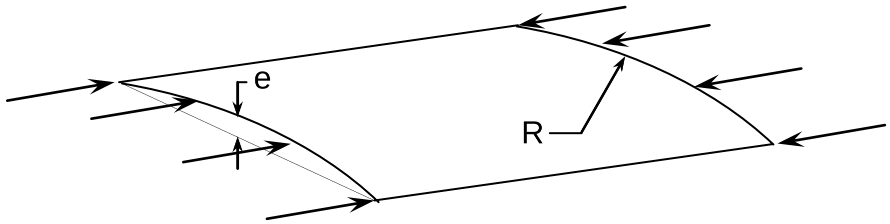


Figure 2.4-12 Definition of Compression Loads Normal to Curvature

2.4.7 Material Allowables

Analysis of composite structures must generate margins of safety based on allowable design values. A design value for a material is the minimum value of a material property expected to be present in production structure. The structural analyst should understand the basis for design values and know where to obtain the design values that should be used in analysis. This section discusses the basis for composite design values and source for composite static and durability and damage tolerance (DaDT) allowables.

2.4.7.1 Basis for Allowables

An allowable design value can be deterministic or statistically based. A deterministic design value implies that material is rejected if sample testing determines its properties fall below the established basis value, usually designated as S-basis. An S-basis design value is the specified minimum value of the appropriate Lockheed Martin, government, or industry (*e.g.*, SAE Aerospace) material specification.

A statistically based design value regards the material property of interest as a random variable that varies from specimen to specimen according to a probability distribution. The property is expected to be above the statistically based design value most of the time. Statistical methods used in the calculation of statistically based composite design values are discussed in Section 2.4.4. A- and B-basis design values are statistically based and defined as follows:

- A-Basis – At least 99 percent of the population of values are expected to equal or exceed the statistically calculated allowable with a confidence of 95 percent.
- B-Basis – At least 90 percent of the population of values are expected to equal or exceed the statistically calculated allowable with a confidence of 95 percent.

In deterministic structural analysis, deterministic and statistically based design allowables are used in the same way—actual, including appropriate safety factors, stresses or strains in the structure can not exceed the material design values. Only statistically-based design allowables can be used with probabilistic methods, which make reliability estimates.

Statistically based design allowables should be used in composite structural analysis. B-basis allowables are typically used for composite structures but the certifying authority must approve the basis used. See Section 2.4.4 for discussion on development of statistically based composite allowables.

2.4.7.2 Static Allowables

Composite material static design allowables should be obtained from the MATUTL tool in the Integrated Detail Analysis Tool suite, IDAT. This tool provides interactive access to engineering material property data for creation, editing, or viewing. MATUTL contains typical strength values, B-basis design allowables, and elastic, environmental, and fracture properties for composite material systems in use at Lockheed Martin Aeronautics that have completed allowables development programs, as discussed in Section 2.4.4, and been released to programs.

The composite properties in MATUTL include allowables for elevated temperatures and saturated moisture content and the tool provides interpolated values and properties between environmental conditions for which test data exist. The MATUTL material database consists of material classes that are a collection of material data items associated with a particular development program. Structural analysts should consult their program structures guidelines or leadership for direction on which material systems are approved for specific program use.

MATUTL is tightly integrated with laminate editors and analysis tools in IDAT, allowing automated access to material properties and allowables. It also offers a carpet-plotting data visualization function that creates plots of laminate stiffness, strength, and certain fracture properties such as open hole tension and compression. Figure 2.4-13 shows an example window from MATUTL giving elastic and allowable properties for a composite material system. Further details about MATUTL can be found in Reference 2-27.

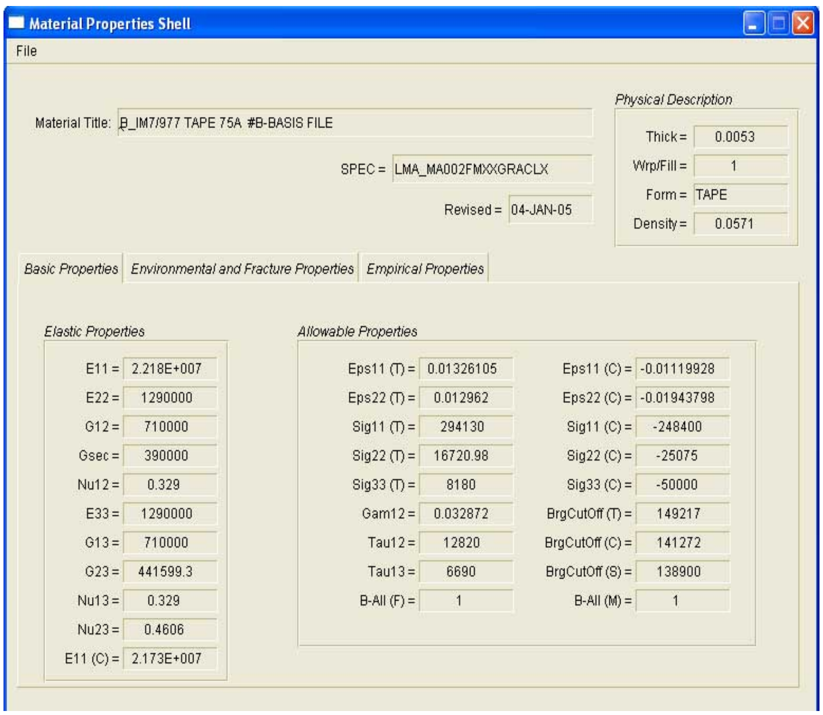


Figure 2.4-13 Composite Material System Elastic and Allowable Properties in IDAT MATUTL

2.4.7.3 Durability and Damage Tolerance Allowables

Durability and damage tolerance design compression and tension strain allowable values for composite material systems are obtained from the IDAT/CDADT composite durability and damage tolerance analysis tool. The CDADT output presents the maximum and minimum principal strains calculated from the defined laminate stacking sequence, load state, and damage assumption. 'Critical' compression and tension 'allowable' strains are also presented and used for margin calculation. Compression and tension strain allowable values are derived from lower bounding of compression and tension after impact tests of composite plates. Damage tolerance strain allowables are not required to represent either statistical A- or B-basis values due to the nature of the empirical derivation and once-per-fleet-life threat. Composite damage tolerance margin calculations are discussed in Section 2.4.2.3 and details of composite damage tolerance analysis methodology are discussed in Section 12.5 of this manual. Further details about CDADT can be found in Reference 2-25.

Composite durability and repairability allowables are determined, as discussed in Section 2.4.2.4, using the IDAT/IBOLT tool for solid laminates and either the IDAT/IBOLT tool or empirical curves for sandwich facesheets.

Repairability is checked using notched allowables in otherwise un-notched composites to allow bolted patch repairs. Details of composite durability and repair analysis methodology are discussed in Section 12.5 of this manual.

2.4.8 Environmental Factors

Composite material allowables are sensitive to environmental effects, particularly temperature and moisture conditions. Analyses of composites should be conducted for the most critical environment applicable to the loading condition. In most instances the critical condition will be hot/wet, but in some cases, such as open hole tension and effects of defects, composite material allowables can be more critical in the cold/dry condition and the part should be analyzed accordingly.

Many issues relating to environmental factors can be avoided through proper design, processing, and handling of composite materials; structural analysts should be aware of relevant material and processing specifications for the material systems in use at Lockheed Martin Aeronautics. These issues include the following:

- Corrosion—Polymer matrix composites are themselves generally insensitive to salt water and many common chemical substances with respect to corrosion. However, chemical paint strippers aggressively attack the polymer matrix of composites and should not be used on composite structure.
- Galvanic corrosion—Carbon fiber is cathodic (noble); aluminum and steel are anodic (least noble). Consequently, carbon in contact with aluminum or steel promotes galvanic action that results in corrosion of the metal. Corrosion barriers (such as fiberglass and sealants) should be placed at interfaces between composites and metals to prevent metal corrosion.
- Ultraviolet radiation—Long term exposure to ultraviolet rays from the sun degrades epoxy resins. This is easily prevented by a surface finish such as a coat of paint.
- Erosion or pitting—Erosion or pitting can be caused by high speed impact with rain or dust particles and is likely to occur on unprotected leading edges. Surface finishes such as rain erosion coats and paints can prevent surface wear.

Some other environmental factors have a more pervasive impact on the analysis of composite structures. The sections that follow discuss the effects on composite materials of the thermal and moisture environment, the vibroacoustic environment, and the electromagnetic environment.

2.4.8.1 Thermal and Moisture

Composite material mechanical properties are strongly influenced by both temperature and absorbed moisture, with combined temperature and moisture having a synergistic effect on properties. Consequently, properties for these materials are typically supplied for specific moisture content at several discrete temperatures spanning the range of aircraft service temperatures. The standard moisture content for design is that corresponding to saturation at 85% relative humidity, or approximately 1.0% by weight for epoxy composite materials. This condition is considered representative of a worst case operational basing scenario of ten years in Guam. The elevated temperature and moisture condition properties for approved composite materials may be obtained from the IDAT/MATUTL tool.

For temperatures between those available in published allowables, interpolation may be used to calculate properties. Extrapolation to estimate properties outside the range of published allowables should not be used. The IDAT/MATUTL tool will interpolate to a given temperature that is between valid operating temperatures. In general, for composites, cold dry conditions generate the critical case for in-plane tension loading. Similarly, hot wet conditions generate the critical case for in-plane compression, shear, and out-of-plane tension loading.

Composite materials are also evaluated for strength retention after exposure to various fluids, thermal spiking, and thermal cycling. These requirements, which are usually program specific, are imposed on materials but, while accounted for in allowables, are not individually evident in allowable values.

2.4.8.2 Vibroacoustic

Composite structures frequently operate in acoustic and vibration environments and analysis must show structural capability to withstand these environments. In general, relative to metallic materials, composite materials typically perform well in vibroacoustic environments and composite components tend to be insensitive to random high-cycle, low-load-amplitude excitation environments. Dynamic analysis of composite structures is beyond the scope of this manual. Analysts should be aware of vibroacoustic environments and consult the Dynamics group for guidance on dynamic analysis of composite structures.

Examples of aeroacoustic load sources include

- Propulsion system noise, exhaust, and temperature
- Cavity noise from any cavity exposed to external flow; *e.g.*, open weapon bays or landing gear bays
- Boundary layer pressure fluctuations arising from high dynamic pressure and transonic flight conditions and separated flows due to protuberances or discontinuity in external surfaces
- Blast pressures due to gunfire and rocket motor firing
- Aeroacoustic loads in ram air ducts, inlets, air conditioning ducts, plenums, and fans and all doors and cavities exposed to external flow
- Motors and pumps

Examples of vibration load sources include

- Forces and moments transmitted to the aircraft structure mechanically or aerodynamically from the propulsion system, jet exhaust and aerodynamic wakes, downwashes and vortices (including those from external stores carriage, speed brakes, forebody strakes, weapons bay doors, antennae, etc.) and cavity resonances
- Forces from gun recoil or gun blast
- Buffeting forces
- Unbalances, both residual and inherent, of rotating components, including the engine and auxiliary power units
- Forces from store carriage and ejection
- Forces due to operation from airfields and ships
- Structural response due to gusts

Analysts should be aware that laminated plate vibration responses are sensitive to the laminate out-of-plane stiffness and, therefore, laminate stacking sequence. Complex interactions among laminate stacking sequence, plate geometry, and boundary conditions preclude simple general rules relating laminate stacking sequence to vibrations. Instead, such rules must be established for specific structure and boundary conditions. Higher fundamental frequencies tend to occur for square plates with preferential stacking of ±45° plies in outer layers. The strongest effect of laminate stacking sequence occurs for rectangular plates in which preferential stacking of outer plies oriented perpendicular to the longest plate dimension have the highest fundamental frequencies.

2.4.8.3 Electromagnetic

Non-conducting composite structures must address the aircraft electromagnetic environment differently than conducting metallic structures. In particular, composite structures can be vulnerable to damage from lightning strikes, allow electrical charge to flow to internal aircraft systems (*e.g.*, wiring, hydraulic lines, fuel tubes, ventilation ducts) and provide less shielding of electronic systems from lightning electromagnetic fields. Certain regions of the aircraft require lightning protection in the form of adequate skin thickness or plies of lightning strike protection material as discussed in Sections 2.3.1 and 2.3.6. In cases where substructure is also composite, the inside ends of attachment bolts may need to be connected with each other and to ground by a conducting wire.

2.5 Special Inspection and Test Requirements

Section 2.9.1 of Reference 2-1 provides a general discussion of inspection and test requirements for aircraft structures. Like metallics, composite materials and structures require testing and inspection at stages throughout the design and manufacturing process to ensure integrity of the finished structure. This section provides an overview of these testing and inspection processes relevant to composite materials and structures. The following aspects of composite testing and inspection are discussed in this section:

- The building block approach used to characterize structural capability for certification
- Qualification of new, or changes to existing, material sources, forms, or processes
- Testing used to calibrate analysis tools to ensure accurate prediction of structural response
- Testing and inspection to ensure quality of the material system and the finished part

2.5.1 Building Block Approach

Composite structures design and development must assess structural performance and durability prior to aircraft certification and operational use. Such assessment consists of a mix of test and analysis. Testing alone is too expensive due to the cost and complexity of the large number of tests required to verify every geometry, loading, environment, and failure mode in the structure. Analysis alone is not sophisticated enough to ensure the reliable prediction of structural response required. A combination of test and analysis, in which analytical predictions are verified by test and testing is guided by analysis, is required in order to minimize cost and ensure reliable prediction of structural performance. This combination of testing and analysis is applied to structural development in progressing levels of structural complexity and is known as the building block approach. This section provides a brief introduction to the building block approach. Further details are discussed in Section 18.4 of this manual.

In the building block approach, as shown in Figure 2.5-1, structural substantiation begins with testing small coupon specimens and progresses in complexity to full-scale product testing. A large number of less complex, low cost specimens are tested early with complexity and cost growing at each level. Only a small number of more complex, high cost full-scale articles are tested. Each level of testing builds on knowledge learned from previous, less complex levels and analysis based on knowledge from less complex testing guides the planning of more complex tests. The typical levels of testing are as follows:

- Coupon tests—define fundamental material system properties, design allowables, and physical characteristics, accounting for sensitivities such as environmental extremes, fatigue, and notches. These material system characteristics are used in structural analysis, analysis tool calibration, and quality assurance.
- Element tests—evaluate simple design details like joints and introduce variables such as damage, manufacturing variability, environment, and fatigue. Element tests provide empirical correlation for strength failure criteria and typically use uniaxial loading.
- Subcomponent tests—evaluate regions of structure for more complex requirements like birdstrike, lightning strike, fuel containment, assessing capability at environmental conditions when required, and address fabrication and quality issues that arise from larger scale structure. Subcomponent tests provide empirical correlation for strength failure criteria and typically incorporate multi-axial loading.
- Component tests—validate designs of structure that can be effectively tested off of full-scale test articles. Examples include horizontal tails, landing gear, and in-flight opening doors.
- Full-scale airframe tests—verify predicted internal load distributions, structural static strength, and service life predictions. Additionally, full-scale structural test articles are used for ground vibration testing, carrier drop testing, and carrier catapult and arrestment testing. Full-scale structural testing is used to provide envelope clearance for flight test programs.

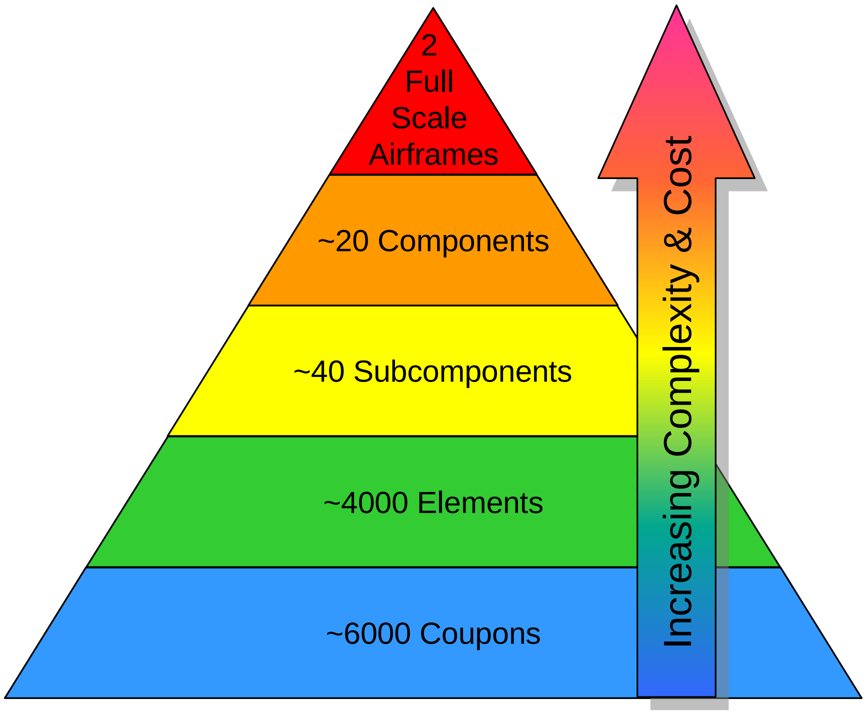


Figure 2.5-1 Example of Progressive Levels of Testing in the Building Block Approach

The building block approach effectively assesses technology risks early in a program lifecycle, reducing program cost and risk while ensuring all technical, regulatory, and customer requirements are met. The approach forms an integral part of the airframe certification plan, encompassing many objectives in support of the structural substantiation required for airframe certification. These objectives include the following:

- Characterization—fundamental material properties and physical characteristics
- Allowables—statistically compensated elastic constants for strength analyses
- Analysis calibration and correlation—empirical input data for analysis tools
- Property modification and effect factors—account for environmental, manufacturing, or design specific effects
- Risk reduction—uncover unknowns and establish confidence early at lower cost
- Qualification—confirmation of an attribute or property
- Certification—tests not otherwise possible on full-scale airframe

2.5.2 Qualification

New composite material systems or processing methods must be qualified for application to aircraft components. Qualification evaluations involve substantial testing through coupon and element levels of the building block approach discussed in Section 2.5.1 using the methods discussed in Section 2.4.4. The qualification effort should account for multiple sources, product forms, and processes to ensure complete characterization of the material system. Prior to start of qualification testing, the material parameters must be fully defined and have a released process specification. If a change is required to a material source, product form, or process, the new source, form, or process must be evaluated for equivalency with material and process specifications in use on aircraft programs. This evaluation involves coupon and element testing sufficient to demonstrate properties equal to or better than the source, form, or process being replaced, across the full range of significant properties.

2.5.3 Analytical Tool Calibration

Analysis of composite structures relies heavily on empirical factors and calibration of analytical models based on test data. Solid mechanics provides the theoretical basis for aircraft structural analysis methods. Theory must be validated by testing and analytical equations must have coefficients and factors determined by test in order to calculate physically meaningful results. Specialized analytical tools, focused on specific analysis, have empirical calibration built in while generalized finite element tools require the analyst to ensure appropriate factors are accounted for in detail models.

2.5.3.1 Embedded Calibration

Composite structural analysis tools in the Lockheed Martin Aeronautics Integrated Detail Analysis Toolset (IDAT) use analytical predictions that are empirically calibrated with test data. Empirical factors in these tools account for various geometric, material, fastener, and configuration variables in the embedded methodologies. Some examples include IBOLT, SFBOLT, CDADT, and adhesive materials used in A4EI.

As discussed in Section 2.4.2.2.1, notched laminate allowables from IBOLT use configuration factors determined from testing that are applied to standard configuration allowables to account for joint configuration details such as diameter, fastener/hole clearance, fastener torque, single/double shear joint, countersunk fastener head, e/D, W/D, percent 0° plies, and percent 90° plies. These factors are contained in the material data files and are divided into three basic categories: basic in-plane factors, bearing cut-off factors, and bending factors. Detailed descriptions of these factors with equations for the calculation of analytical factors are contained in Appendix A of the IBOLT Theory Manual, Reference 2-35. Further details about IBOLT can be found in Reference 2-22.

Another tool, SFBOLT, calculates fastener static and fatigue margins of safety based on a set of assumptions. SFBOLT fatigue analysis relies on a test database to empirically derive modification factors to the allowable stress based on joint configuration parameters. These modification factors are based on test data of various joint configurations, including variations of metallic and composite materials, laminate stacks, and fasteners. Further details about SFBOLT can be found in Reference 2-36.

The composite DADT program (CDADT), discussed in Section 2.4.2.3, calculates margins for compression after impact (CAI) and tension after impact (TAI) using equations developed from test data. Further details about CDADT can be found in Reference 2-25.

The A4EI bonded joint analysis tool uses reduced adhesive allowables in order to empirically account for adherend interlaminar failure modes that are observed in testing. A4EI uses a 1-D displacement-compatibility model including elastic perfectly plastic adhesive behavior and linear-elastic adherend response but does not account for peel. It predicts shear yield and failure loads for the adhesive and tensile or compressive net section failure of the adherends. Since A4EI was originally intended for analysis of metallic bonded joints, the composite-to-composite bonded joint analysis does not explicitly check the composite adherends; rather it uses reduced adhesive allowables in order to empirically account for adherend interlaminar failure modes based on test results. Further details about A4EI can be found in Reference 2-37.

While empirically calibrated analysis tools ensure good prediction of structural performance within the range of tested configurations, the consequence of this calibration is a constraint on the range of applicability of the tool. The range of analysis validity is limited to the range of test data, which usually spans the range of reasonable usage. The analyst must exercise good engineering judgment when using tools outside range of test data used to calibrate a tool. Tools in the IDAT toolset typically provide a warning when input parameters are near or exceed acceptable ranges.

2.5.3.2 General Model Calibration

Analysis of composite structures using generalized tools such as finite element analysis should also have a sound basis in physically meaningful test results. Meshing, boundary conditions, load application, internal connections, carve-outs, and other idealization methodologies should follow appropriate finite element modeling standards that have been shown to reproduce physical test response in terms of deflection, strain, and failure modes.

In particular, finite element methods used to predict composite failure modes must be correlated with element test data to ensure accurate physical representation of local stresses, strains, and deformation at the initial damage event and thus accurate failure prediction. Analysis of composite detail features like chevron notches, joggles, stiffeners, tight radii, and cutouts frequently relies on load data generated from finite element models. Failure in such composite features typically occurs at geometric stress risers and can sometimes involve induced interlaminar stresses, which are a weakness for composite materials. Consequently, the analyst must apply mesh refinement in these regions sufficient to ensure converged finite element results, which should be verified through convergence studies or alternate method such as non-averaged element fringe plots. Critical strains used to check the detail failure modes in these features must be extracted at a location, or characteristic distance, that corresponds to the extraction location and mesh densities used for test correlation. The characteristic dimension is defined as the distance from the edge of a stress singularity to a point at which the unnotched fiber strain allowable can be applied. Such data should be provided in program guidance such as Reference 2-11.

2.5.4 Quality Assurance

Ongoing integrity of a characterized material system requires testing to ensure quality of each lot of material. This testing must economically and efficiently assess the critical properties of each lot of material to ensure integrity of material properties and fabrication capability and conformance to specification standards. Production quality control involves inspection and testing of composites in all stages of manufacture and part fabrication. Tests are performed by the material supplier on fiber and resin as separate materials as well as on composite prepreg material. Additionally, Lockheed Martin Aeronautics performs receiving inspection and revalidation tests, in-process inspection, and nondestructive inspection on finished parts. This inspection and testing is described in the following sections.

Lockheed Martin Aeronautics controls composite part quality through material and process specifications. Material specifications control the purchase of structural materials to ensure consistency over time of material quality and properties. In this way structural design, analysis, and manufacturing can proceed with confidence that established allowables and properties, developed through expensive qualification testing, will be achieved in finished products. Material specifications are frequently specified in contract terms and are part of purchase order requirements to procure material. Process specifications establish procedures required to control the finished product. Because composite fabrication and adhesive bonding processes are sensitive to process variation and must satisfy rigorous finished product requirements, the associated process specifications are typically very detailed. Structural analysts should be familiar with the material and process specifications applicable to their program.

The discussion of testing and inspection in this section is for reference only in order to familiarize the analyst with industry practices that could be encountered. Materials and Processes and Quality Assurance personnel perform most of the testing and inspections discussed in this section and should always be consulted in matters concerning test and inspection. In all cases, production work instructions and material and process specifications take precedence over this discussion.

2.5.4.1 Receiving Inspection

Lockheed Martin Aeronautics material specifications define incoming material inspection procedures and supplier controls that ensure consistent quality of the materials used in composite manufacturing so that the finished product performs as designed. These specifications call out acceptance criteria based on mechanical tests that assure critical material properties for each lot of material to be used in manufacturing are equivalent to the material used to develop allowables. Material specifications require suppliers to provide test data, certification, affidavits, and other evidence, depending on contract requirements, that each production lot of material shipped meets the material specification requirements.

Acceptance tests must be sufficient in scope to assure incoming material will meet or exceed engineering requirements established during material qualification. As an example, Table 2.5-2, reproduced from Reference 2-7, lists some possible acceptance tests that might be required for prepreg tape material. The first part of this table lists testing that could be done on uncured prepreg to ensure resin and fiber materials are conform to standards. The second part of the table lists testing that could be done on cured laminates to ensure mechanical properties of cured material meet specification requirements. Mechanical property acceptance tests cover fiber and resin properties in tension, compression, and shear and include some elevated temperature testing of resin dependent properties. These tests can be direct tests of a property or a basic test that correlates with critical design properties. Specific Lockheed Martin material specifications indicate required acceptance tests, who performs the test (Lockheed Martin or the vendor), the frequency of each test, and, in the event of initial failure to satisfy these requirements, retest criteria.

Table 2.5-1 Possible Acceptance and Revalidation Tests (Reproduced from Reference 2-7)

Property	Testing required			Specimens Required Per Sample
	Production Acceptance (Supplier)[3]	Production Acceptance (User)[3]	Revalidation (User)[3]	
Prepreg Properties				
Visual & Dimensional	X	X		-
Volatile Content	X	X		3
Moisture Content	X	X	X	3
Gel Time	X	X	X	3
Resin Flow	X	X	X	2
Tack	X	X	X	1
Resin Content	X	X		3
Fiber Areal Weight	X	X		3
Infrared Analysis	X			1
Liquid Chromatograph	X	X	X	2
Differential Scanning Calorimetry	X	X	X	2
Lamina Properties				
Density	X			3
Fiber Volume	X			3
Resin Volume	X			3
Void Content	X			3
Per Ply Thickness	X	X	X	1
Glass Transition Temp	X	X	X	3
SBS or ±45º Tension	X[2]	X[2]	X[2]	6
90º/0º Compression Strength	X[1]	X[1]	X[2]	6
90º/0º Tension Strength & Modulus	X[2]	X[2]	X[2]	6

[1] Tests should be conducted at RT/Ambient and Maximum Temperature/Ambient
[2] Tests should be conducted at RT/Ambient
[3] Supplier is defined as the prepreg supplier. User is defined as the composite part fabricator. Production acceptance tests are defined as tests to be performed by the supplier or user for initial acceptance. Revalidation tests are tests performed by the user at the end of guaranteed storage life or room temperature out time to provide for additional use of the material after expiration of the normal storage or out time life.

2.5.4.2 In-process Inspection and Testing

Composite manufacturing process verification requires control and tracking of all materials, operations, equipment, and facilities involved in fabricating composite parts. Lockheed Martin process specifications define systems for controlling and tracking composite manufacturing processes, and structural analysts should be familiar with the specifications applicable to their program. To ensure quality and integrity in finished composite parts, certain critical operations must be closely controlled during the fabrication process. Process specifications call out requirements and limits for these operations.

Operations that must be closely controlled include the following:

- Verification that release agent has been applied and cured on a clean tool surface
- Verification that perishable materials used in the part comply with applicable material specifications
- Inspection of lay-ups to verify ply count, orientation, and sequence comply with engineering drawing requirements
- Inspection of honeycomb core installation to verify positioning complies with engineering drawing requirements
- Documentation of the following:
  - Material supplier, date of manufacture, batch number, roll number, and total accumulated hours of working life
  - Autoclave or oven pressure, part temperatures, and times
  - Autoclave or oven load number
  - Part and serial number

Statistical process control (SPC) methods are typically employed for many of these fabrication steps. Discussion of SPC is beyond the scope of this manual and the structural analyst should consult with production operations and quality assurance personnel if further information regarding SPC is required to support structural integrity tasks.

Parts for critical applications or that are difficult to inspect may require special process control test panels, often called travelers, to be fabricated and cured along with production parts. These panels are then tested after cure to verify physical and mechanical properties meet engineering requirements for the parts they represent. The engineering drawing typically calls out requirements for physical and mechanical testing by designating a type or class for each part. Whereas non-critical or secondary structure may require no test specimens and no testing, critical or safety-of-flight parts may require complete physical and mechanical testing. Process control specimen testing requirements could be limited to a few mechanical property tests such as 0° flexure strength and modulus and short beam shear strength or they could include more extensive physical property testing such as glass transition temperature, per ply thickness, fiber volume, void content, and ply count on samples taken from designated areas on a sample production part.

2.5.4.3 Nondestructive Inspection

Quality assurance requires that detail composite parts are inspected for conformance to dimensional and workmanship requirements to ensure manufacturing-induced defects and damage fall within established limits. Reference 2-1 Section 2.9.1 provides a general discussion of nondestructive inspection (NDI) techniques used on aircraft structures. Section 18.4 of this manual provides further details on NDI in the context of certification of composite structures.

Laminates are prone to particular types of defects unless they are properly assembled, cured, machined, drilled, and otherwise handled. Lockheed Martin Aeronautics process specifications establish workmanship standards for composite fabrication processes and control the quality of manufacturing operations on composite parts. These specifications establish visual acceptance and rejection limits for typical defects such as

- Splintering
- Delamination,
- Loose surface fibers
- Overheating
- Surface finish
- Off-axis holes
- Surface cratering

Many defects in composite parts occur internally and therefore are not able to be assessed for acceptance through visual inspection. These internal defects can substantially affect bearing, compression, and interlaminar strengths so evaluation of the seriousness of these flaws is critical. Consequently, NDI techniques that can identify internal defects must also be used to ensure composite part quality. Lockheed Martin Aeronautics process specifications establish acceptance and rejection limits for internal defects such as the following:

- Delaminations
- Impact damage
- Hole breakout
- Porosity
- Voids
- Inclusions

The extent of required NDI on composite parts depends on the part classification, which was discussed in Section 2.2. In particular, Aircraft Structural Integrity Program (ASIP) classification designations specify inspection levels and intervals throughout the lifecycle of certain parts. Safety-of-flight and other selected structural components of the airframe must be capable of maintaining required residual strength in the presence of material, manufacturing, and processing defects, and damage induced during normal usage and maintenance until the damage is detected through periodic scheduled inspections. The engineering data package identifies the type or class of a part and references a process specification, which defines the NDI tests and the accept/reject criteria.

References 2-26 and 2-34 provide guidance and detailed information on NDI methods and processes at Lockheed Martin Aeronautics. Composite structure details such as skins, edge members, and laminates are typically inspected using ultrasonic or optical methods such as through-transmission, pulse-echo, resonance, laser ultrasonics, infrared thermography, and shearography. Additional inspection methods may be used based on flaw detection capabilities or inspection difficulties. Common NDI techniques are briefly discussed below under the headings of visual, ultrasonic, and radiographic inspection.

2.5.4.3.1 Visual Inspection

Visual inspection checks to assure parts meet drawing requirements and evaluates the surface and appearance of the part. The inspection includes examination for blisters, depressions, foreign material inclusions, ply distortions and folds, surface roughness, surface porosity, and wrinkles. Accept/reject criteria for such defects are given in Lockheed Martin Aeronautics process specifications. Additionally, visual inspection can give insight into issues not covered by the specification.

2.5.4.3.2 Ultrasonic Inspection

The ultrasonic test method is a volumetric inspection of the test component that can detect porosity, foreign material inclusion, voids, delaminations, thickness changes, and changes in geometry. Ultrasonic testing covers a broad range of methods, including the basic tap test and conventional and laser transmission media. The simplest sonic inspection technique involves striking the surface of the object being evaluated in multiple places with a tap hammer and listening to the sound. As the tapping passes over an indication, the sound will change from a ring to a thud. This inspection technique is best performed in a quiet place by someone with good hearing. While limited in sensitivity and not typically used in a production environment, this technique is sometimes the only one available for inspecting an aircraft.

In conventional ultrasonic testing the ultrasound is produced by a piezoelectric transducer. The energy produced by the transducer is transmitted through a couplant (water, water column, or water-soluble gel) directly into the test component. The ultrasonic energy transmitted through the test component is then measured as amplitude of the reflected signal (Reflector Plate) or transmitted energy (Through-Transmission Ultrasonic), or the amplitude and transit time of the reflected signal (Pulse Echo or Shear Wave). Ultrasonic inspection can be performed using an automated system like an immersion tank, a water column system, or a hand scan unit (where the qualified operator scrubs the surface of the component with a transducer).

In laser ultrasonic technology, ultrasound is induced in the composite material by the absorption of energy from a pulsed generation laser, essentially making the composite surface the transducer.

It is generally necessary to have an ultrasonic inspection standard, referred to as an Inspection Reference Panel (IRP), representative of the geometry and material of the composite structure to be inspected. The IRP is a manufactured standard with implanted flaws or flat bottom holes at known depths. Programs may require a zone diagram on the face of the drawing identifying the inspection requirements. This diagram is used by NDI personnel to determine the necessary IRPs and to develop the Technique Data Sheet (TDS), which provides the inspector with the information, such as inspection technique, location, IRP Number, etc., necessary to conduct the inspection. The inspection criteria for composite parts are program specific and are jointly determined by Materials and Processes, Manufacturing, Quality Assurance, NDI, Design, and Analysis engineers.

Ultrasonic inspection data can be presented in several forms, which include A-scan, B-scan, and C-scan. A-scan presentation displays the detected signal amplitude versus time. As shown in Figure 2.5-2, the horizontal axis on the A-scan chart indicates elapsed time and the vertical axis is amplitude. A-scans offer a great deal of information regarding size, depth, and type of defect. Normally, A-scan inspection does not record waveform data.

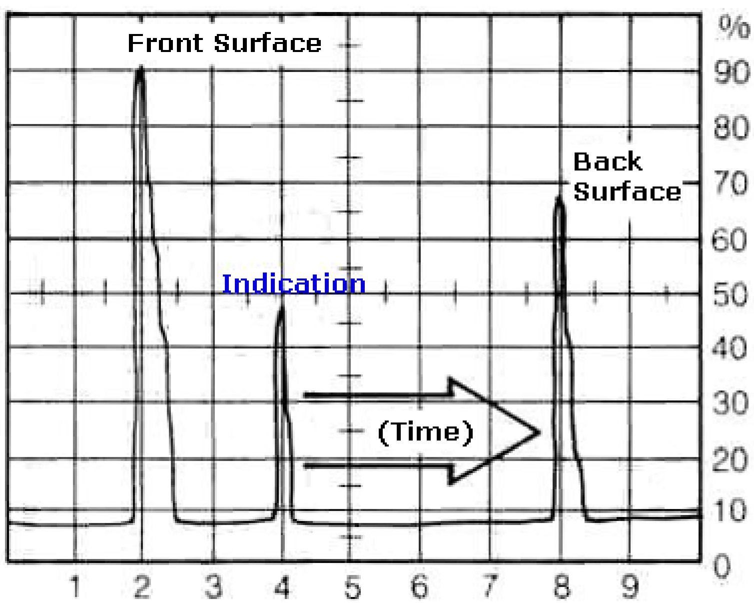
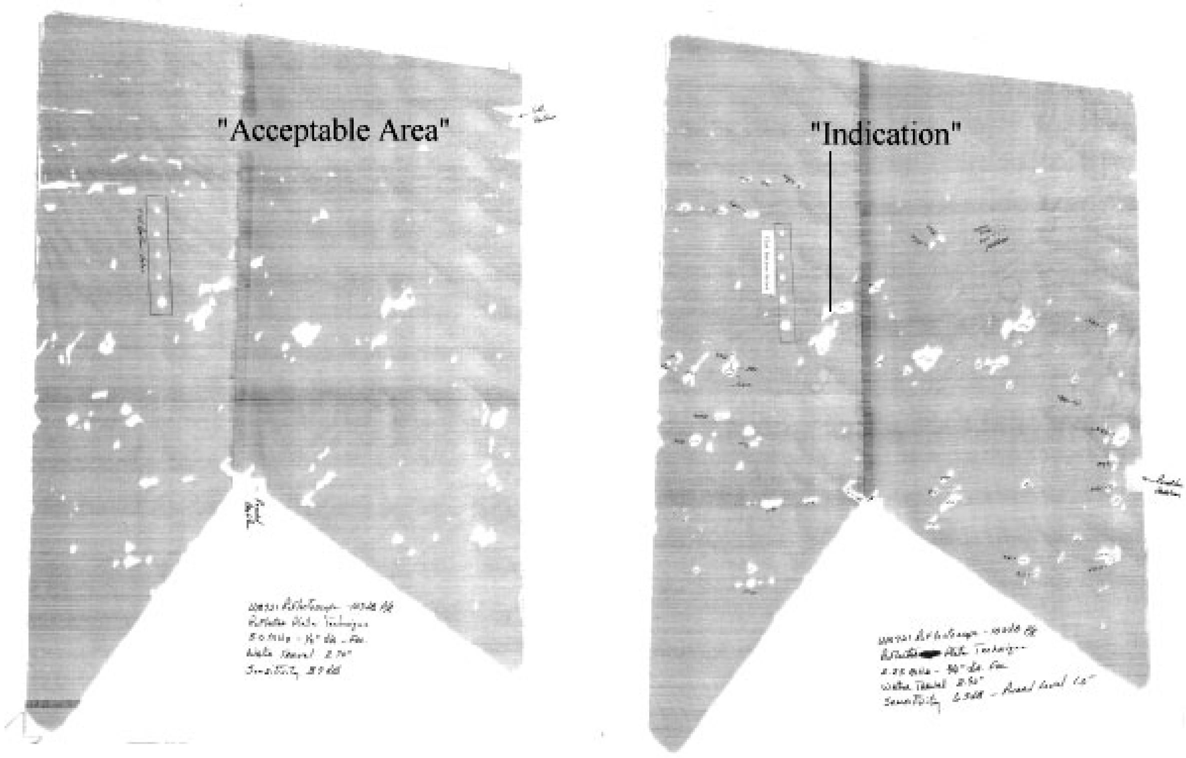


Figure 2.5-2 Example of A-scan Presentation of Results

B-scan presentation is a cross-sectional view of the material under inspection. The B-scan, via computer monitor or paper recording, displays the front surface, back surface, and any discontinuities between the front and back surfaces.

C-scan presentation displays the inspection results in a plan view. It can display results of changes in amplitude or transit time of the reflected signal. C-scans can be recorded on computer monitor or paper, and are extremely useful as inspection presentations for automated equipment. Figure 2.5-3 shows ultrasonic inspection data presented as a C-scan.



5.0MH$_z$ and 2.25MH$_z$ Reflector Plate C-Scan Without Overlay.

Figure 2.5-3 Example of C-scan Presentation of Inspection Results

2.5.4.3.3 Radiographic Inspection

Radiographic, or X-ray, inspection utilizes the penetrating power of the projected X-ray radiation to reveal the internal nature of test object. The X-ray beam passes through the test object and strikes a film, creating a latent image. The projected radiation striking the film is absorbed to varying degrees dependent upon the radiation energy, material density, atomic number and the thickness variations of the material. As with ultrasonic inspection, standards with built-in defects are usually required to evaluate the radiographic film properly. Radiographic inspection is frequently used to evaluate bonding of inserts in laminate panels and honeycomb core to facesheet bonds in sandwich panels.

Radiographic inspection has several advantages. The method can
- Reveal the internal condition of most materials
- Reveal fabrication and assembly error
- Provide a permanent visual image of the test object in X-ray film
- Detect in the structure a 2% density variation, which is based on the film and the ability of the human eye to detect very small changes in the film density

Radiographic inspection also has several disadvantages. The method
- Requires two-sided accessibility
- Will not detect all discontinuities—depends on size, orientation, and type
- Is expensive in both time and equipment costs.
- Presents a potential health hazard due to radiation generated

The sensitivity of radiographic inspection for locating problems in core-bonded assemblies is very good but the method has limits when it comes to inspecting a vertical core splice (core-to-core or core-to-detail). The width of a separation or a crack in the adhesive can be observed but the depth of the separation cannot be determined and must therefore be assumed to be completely through the assembly. The sensitivity of the method can be enhanced by use of image enhancing penetrants but their use can lead to other problems. It is not recommended that solid laminates be routinely X-rayed; technical leadership approval should be obtained for drawing callout of radiographic inspection on a production laminate.

2.5.4.3.4 Preferred NDI Methods

Reference 2-26 provides guidance on preferred methods for detecting typical flaws in composite parts as shown in Table 2.5-1.

Table 2.5-2 Ranking of NDI Methods for Composites Flaw Detection

Flaw Type	**TTU**	**PE**	**SW**	**X-Ray**	**Tap Test**
Porosity	1	1	2	1	
Foreign Material	2	1		2	
Shallow Delaminations	1	1			1
Deep Delaminations	1	1			
Matrix Cracks			1	1	
Fiber Breaks				1	
Impact Damage	1	1	2		2
TTU = Through-Transmission Ultrasonic, PE = Pulse-Echo Ultrasonic, SW = Shear Wave Ultrasonic, X-Ray = Radiography, Tap Test = Tap Hammer					
1. Provides significant detection reliability in most cases. Good candidate for primary inspection method. 2. Provides somewhat reduced detection reliability, or may be applicable to a limited condition. May be a good candidate for backup or special purpose application.					

2.5.4.4 Destructive Testing and First Article Inspection

Destructive tests are used to ensure structural integrity when nondestructive techniques alone cannot ensure quality of a finished composite part. Such testing can be either full part dissection or trim sections cut from excess parts of the component. In either technique, inspections are performed for a variety of concerns; for example, to verify ply stacking sequence, to examine microstructure using photomicrographs, or to investigate ply wrinkles and porosity. Test coupons are cut from full parts or trim sections and tested for the critical failure mode for the area of the part where the specimen was cut. Typical failure modes tested include unnotched compression, open hole compression, and interlaminar tension and shear. Destructive testing generally cannot verify historical or building block test results because part sections do not match previous test coupon configurations. Rather, mechanical testing can be used to assess analysis methodology and predictions. The analyst must request destructive tests and define the specific feature(s) to be observed in a cut specimen.

Full part dissection can give a complete examination of the part. Sections and specimens from a production part are cut, inspected, tested, or otherwise examined. Full dissection is often performed on the first part from a new tool. Because it prevents any future use of the part, full part dissection is expensive and should only be used when the following are true:

- Important areas cannot be adequately inspected using nondestructive methods
- The part is complex
- Low experience level with part configuration or fabrication process
- The part is net trim, *i.e.*, excess trim areas or part extensions are not available or suitable for examination

Examination of excess trim sections is the preferred approach whenever possible because the part is not destroyed, structural details can still be examined, and mechanical test specimens can be obtained. Inspection and test specimens come from part extensions intentionally designed to go beyond the trim line or from cutout areas inside the part. These specimens are used for inspection of detail section cuts and mechanical and physical test coupons to ensure structural capability and manufacturing process quality.

Potential areas for destructive testing and inspection include

- Part primary load paths
- Areas that showed NDI indications
- Tool markoff near cocured details
- Ply drop-offs near tapers
- Ply wrinkles
- Resin-starved and resin-rich areas
- Corner radii
- Cocured details
- Core to face sheet fillets
- Tapered core areas

Periodic destructive testing and inspection monitors manufacturing processes to ensure part quality, demonstrate process capability, and identify ranges of suspect parts. The program fracture control plan drives destructive testing requirements, including frequency. The frequency of destructive tests depends on part criticality, part complexity, and fabrication process experience. Safety-of-flight or otherwise critical parts may require ongoing destructive testing at regular intervals. Parts with complex features or other fabrication challenges will likely require destructive testing at frequent intervals until process capability is demonstrated, at which point the intervals may lengthen. Destructive testing is typically performed on the first part from a new tool or from a new vendor.

Less elaborate trim section testing and inspection can be executed more frequently and at less cost than full part dissection and should be considered when possible. Parts may be sampled to demonstrate process capability without the need for 100% testing. Sampling typically includes full dissection of the first article followed by periodic trim section test and inspection, with the frequency depending on success rate and part criticality. Because of the lower cost, periodic destructive tests on trim sections can be conducted at shorter intervals than full dissection and may therefore be preferred for critical parts.

2.6 Content of Stress Analysis Reports

The content of stress analysis reports is covered in Section 2.10 of Reference 2-1. The analyst should ensure that all relevant composite failure modes are addressed in the stress analysis report.

2.7 Errata in Standard Texts

Refer to the errata page on the LMASAM Structural Analysis Website.