CHAPTER 12 LESSONS LEARNED

12.1 INTRODUCTION

The focus of much of what is in this handbook concentrates on establishing proper techniques for development and utilization of composite material property data. The motivation prompting specific choices is not always evident. This chapter provides a depository of knowledge gained from a number of involved contractors, agencies, and businesses for the purpose of disseminating lessons learned to potential users who might otherwise repeat past mistakes. Many of the contractors involved in developing the lessons learned are aerospace oriented. Thus, the lessons learned may have a decidedly aerospace viewpoint.

The chapter starts with a discussion of some of the characteristics of composite materials that makes them different from metals. These characteristics are the primary cause for establishing the methods and techniques contained in the handbook.

Specific lessons learned are defined in later sections. They contain the specific "rule of thumb" and the reason for its creation or the possible consequence if it is not followed. The lessons learned are organized into six different categories for convenience.

12.2 UNIQUE ISSUES FOR COMPOSITES

Composites are different from metals in several ways. These include their largely elastic response, their ability to be tailored in strength and stiffness, their damage tolerance characteristics, and their sensitivity to environmental factors. These differences force a different approach to analysis and design, processing, fabrication and assembly, quality control, testing, and certification.

12.2.1 Elastic properties

The elastic properties of a material are a measure of its stiffness. This property is necessary to determine the deformations that are produced by loads. In composites, the stiffness is dominated by the fibers; the role of the matrix is to prevent lateral deflections of the fibers and to provide a mechanism for shearing load from one fiber to another. Continuous fiber composites are transversely isotropic and in a two-dimensional stress state require four elastic properties to characterize the material:

Modulus of elasticity parallel to the fiber, E_1
Modulus of elasticity perpendicular to the fiber, E_2
Shear modulus, G_{12}
Major Poisson's ratio, ν_{12}

In general, material characterization may require additional properties not defined above. A thorough discussion of this subject is given in Section 5.3.1. Only two elastic properties are required for isotropic materials, the modulus of elasticity and Poisson's ratio.

The stress-strain response of commonly used fiber-dominated orientations of composite materials is almost linear to failure although some glasses and ceramics have nonlinear or bilinear behavior. This is contrasted to metals that exhibit nonlinear response above the proportional limit and eventual plastic deformation above the yield point. Many composites exhibit very little, if any, yielding in fiber dominated behavior. Toughened materials and thermoplastics can show considerable yielding, particularly in matrix dominated directions. This factor requires composites to be given special consideration in structural details where there are stress risers (holes, cutouts, notches, radii, tapers, etc.). These types of stress risers in metal are not a major concern for static strength analysis (they do play a big role in durability and damage tolerance analysis, however). In composites they must be considered in static strength analysis.

In general, if these stress risers are properly considered in design/analysis of laminated parts, fatigue loadings will not be critical.

Another unique characteristic of composite material elastic response is its orthotropy. When metals are extended in one direction, they contract in the perpendicular direction in an amount equal to the Poisson's ratio times the longitudinal strain. This is true regardless of which direction is extended. In composites, an extension in the longitudinal (1 or x) direction produces a contraction in the transverse direction (2 or y) equal to the "major" Poisson's ratio, ν_{xy}, times the longitudinal extension. If this is reversed, an extension in the transverse direction produces a much lower contraction in the longitudinal direction. In fiber dominated laminates, Poisson's ratio can vary from <0.1 to >0.5.

The most unusual characteristic of composites is the response produced when the lay-up is unbalanced and/or unsymmetric. Such a laminate exhibits anisotropic warping characteristics. In this condition an extension in one direction can produce an in-plane shear deformation. It can also cause an out-of-plane bending or torsional response. All these effects are sometimes observed in one laminate. This type of response is generally undesirable because of warping or built-in stresses that occur. Hence, most laminate configurations are balanced and symmetric.

Classical lamination theory is used to combine the individual lamina properties to predict the linear elastic behavior of arbitrary laminates. Lamination theory requires the definition of lamina elastic properties, their orientation within the laminate, and their stacking position. The process assumes plane sections remain-plane and enforces equilibrium. Lamination theory will solve for the loads/stresses/strains for each lamina within the laminate at a given location for a given set of applied loads. This combined with appropriate failure theory will predict the strength of the laminate (empirically modified input ply properties are often necessary).

12.2.2 Tailored properties and out-of-plane loads

The properties of a composite laminate depend on the orientation of the individual plies. This provides the engineer with the ability to tailor a laminate to fit a particular requirement. For high axial loads predominantly in one direction, the laminate should have a majority of its plies oriented parallel to that loading direction. If the laminate is loaded mostly in shear, there should be a high percent of ±45° pairs. For loads in multi-directions, the laminate should be quasi-isotropic. An all 0° laminate represents the maximum strength and stiffness that can be attained in any given direction, but is impractical for most applications since the transverse properties are so weak that machining and handling can cause damage. Fiber-dominated, balanced and symmetric, laminate designs that have a minimum of 10% of the plies in each of the 0°, +45°, -45°, and 90° directions are most commonly used.

Tailoring also means an engineer is not able to cite a strength or stiffness value for a composite laminate until he knows the laminate's ply percentages in each direction. Carpet plots of various properties vs. the percent of plies in each direction are commonly used for balanced and symmetric laminates. An example for stiffness is shown in Figure 12.2.2. Similar plots for strength can also be developed.

Out-of-plane loads can also be troublesome for composites. These loads cause interlaminar shear and tension in the laminate. Interlaminar shear stress can cause failure of the matrix or the fiber-matrix interphase region. Interlaminar shear and tensile stresses can delaminate or disbond a laminate. Such loading should be avoided if possible. Design situations that tend to create interlaminar shear loading include high out-of-plane loads (such as fuel pressure), buckling, abrupt changes in cross-section (such as stiffener terminations), ply drop-offs, and in some cases laminate ply orientations that cause unbalanced or unsymmetric lay-ups. Interlaminar stresses will arise at any free edge. Interlaminar stresses will arise between plies of dissimilar orientation wherever there is a gradient in the components of in-plane stress.

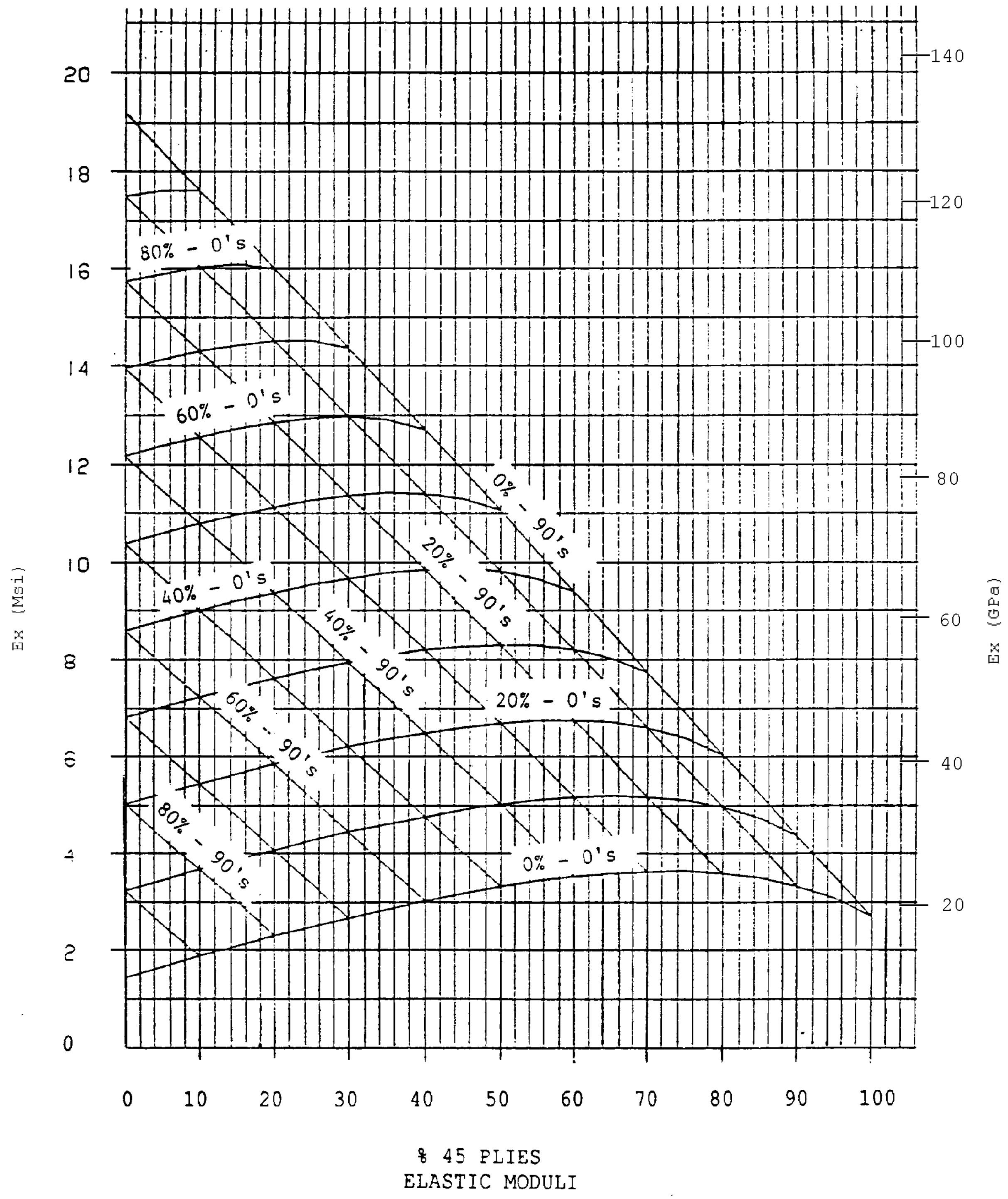


FIGURE 12.2.2 *Sample carpet plot.*

12.2.3 Damage tolerance

Damage tolerance is the measure of the structure's ability to sustain a level of damage or presence of a defect and be able to perform its operating functions. The concern is with the damaged structure having adequate residual strength and stiffness to continue in service safely: 1) until the damage can be detected by scheduled maintenance inspection and repaired, or 2) if the damage is undetected, for the remainder of the aircraft's life. Thus, safety is the primary goal of damage tolerance. Both static load and durability related damage tolerance must be interrogated experimentally because there are few, if any, accurate analytical methods.

There are basically two types of damage that are categorized by their occurrence during the fabrication and use of the part, i.e., damage occurring during manufacturing or damage occurring in service. It is hoped that the occurrence of the majority of manufacturing associated damage, if beyond specification limits, will be detected by routine quality inspection. Nevertheless, some "rogue" defects or damage beyond specification limits may go undetected. Consequently, their occurrence must be assumed in the design procedure and subsequent testing (static and fatigue) performed to verify the structural integrity.

Service damage concerns are similar to those for manufacturing. Types of service damage include edge and surface gouges and cuts or foreign object collision and blunt object impact damage caused by dropped tools or contact with service equipment. A level of non-detectable damage should be established and verified by test that will not endanger the normal operation of the aircraft structure for two lifetimes. A certain level (maximum allowed) damage that can be found by inspection should be defined such that the vehicle can operate for a specified number of hours before repair or replacement at loads not exceeding design limit. This damage should also be tested (statically and in fatigue) to verify the structural integrity.

Delaminations can also be critical defects. However, unless they are very large, historically more than 2 inches (50 mm) in diameter, the problem is mostly with thin laminates. Effects of manufacturing defects such as porosity and flawed fastener holes that are slightly in excess of the maximum allowable are usually less severe. They are generally accounted for by the use of design allowable properties that have been obtained by testing specimens with stress concentrations, e.g., notches. Most commonly these are specimens with a centered hole. Open holes are typically used for compression specimens while either open or filled holes (holes with an installed fastener) are used for tension testing. (Open holes are more critical than filled holes for compression. Filled holes may be more critical in tension, especially for laminates with ply orientations with a predominate number of plies in the load direction.) Consequently, the design allowables thus produced may be used to account for a nominal design stress concentration caused by an installed or missing fastener, at least to a 0.25 inch (6.4 mm) diameter, as well as accounting for many other manufacturing defects. This is sometimes called the "rogue flaw" approach to laminate design, see Reference 12.2.3.

12.2.4 Durability

Durability of a structure is its ability to maintain strength and stiffness throughout the service life of the structure. A structure must have adequate durability when subjected to the expected service loads and environment spectra to prevent excessive maintenance, repair, or modification costs over the service life. Thus, durability is primarily an economic consideration.

Metallic structure can be very sensitive to durability issues; major factors limiting life are corrosion and fatigue. Metal fatigue is dictated by the number of load cycles required to start a crack (crack initiation) and the number of load cycles for the crack to grow to its critical length, reaching catastrophic failure (crack growth). Crack/damage growth rate is very dependent on the concentration of stress around the crack.

In composites, it has been demonstrated that one of the most common damage growth mechanisms is intercracking (delamination). This makes composites most sensitive to compression-dominated fatigue loading. A second common fatigue failure mode is fastener hole wear caused by high bearing stresses.

In this failure mode the hole gradually elongates. The most serious damage to composite parts is low velocity impact damage which can reduce static strength, fatigue strength, or residual strength after fatigue. Again, testing is a must!

The strain level of composites in most actual vehicle applications to date has been held to relatively low values. Composites under in-plane loads have relatively flat stress-life (S-N) curves with high fatigue thresholds (endurance limits). These two factors combined have resulted in insensitivity to fatigue for most load cases. However, the greater variability found with composites requires an engineer to still characterize the composite's fatigue life to failure to correctly characterize its fatigue scatter.

12.2.5 Environmental sensitivity

When a composite with a polymeric matrix is placed in a wet environment, the matrix will absorb moisture. The moisture absorption of most fibers used in practice is negligible; however, aramid fibers (e.g., Kevlar) absorb significant amounts of moisture when exposed to high humidity. The absorption of moisture at the interface of glass/quartz fibers is a well-known degrading phenomena.

When a composite has been exposed to moisture and sufficient time has elapsed, the moisture concentration throughout the matrix will be uniform. A typical equilibrium moisture content for severe humidity exposure of common epoxy composites is 1.1 to 1.3 percent weight gain. The principal strength degrading effect is related to a change in the glass transition temperature of the matrix material. As moisture is absorbed, the temperature at which the matrix changes from a glassy state to a viscous state decreases. Thus, the strength properties decrease with increasing moisture content. Current data indicate this process is reversible. When the moisture content is decreased, the glass transition temperature increases and the original strength properties return. With glass/quartz fibers there is additional degradation at the interface with the matrix. For aramid fibers there is additional degradation at the interface with the matrix and, also, in the fibers.

The same considerations also apply for a temperature rise. The matrix, and therefore the lamina, loses strength and stiffness when the temperature rises. This effect is primarily important for the matrix-dominated properties. Temperature rise also worsens the fiber/matrix interface degradation for glass/quartz fibers and aramid fibers. The aramid fiber properties are also degraded by a rise in temperature.

The approach for design purposes is to assume a worst case. If the material is assumed to be fully saturated and at the maximum temperature, material allowables can be derived for this extreme. This is a conservative approach, since typical service environments do not generate full saturation for most complex structures. Once the diffusivity of a composite material is known, the moisture content and through the thickness distribution can be accurately predicted by Fickian equations. This depends on an accurate characterization of the temperature-humidity service environment.

Thermal expansion characteristics of common composites, like carbon/epoxy, are quite different from metals. In the (0 or 1) longitudinal direction, the thermal expansion coefficient of carbon/epoxy is almost zero. Transverse to the fiber (90 or 2 direction), the thermal expansion is the same magnitude as aluminum. This property gives composites the ability to provide a dimensionally stable structure throughout a wide range of temperatures.

Another feature of composites that is related to environment is resistance to corrosion. Polymer matrix composites (with the exception of some carbon/bismaleimides) are immune to salt water and most chemical substances as far as corrosion sensitivity. One precaution in this regard is galvanic corrosion. Carbon fiber is cathodic (noble); aluminum and steel are anodic (least noble). Thus carbon in contact with aluminum or steel promotes galvanic action which results in corrosion of the metal. Corrosion barriers (such as fiberglass and sealants) are placed at interfaces between composites and metals to prevent metal corrosion. Another precaution regards the use of paint strippers around most polymers. Chemical paint strippers are very powerful and attack the matrix of composites very destructively. Thus, chemical paint stripping is forbidden on composite structure.

Other environmental effects worth noting include the effect of long term exposure to radiation. Ultraviolet rays from the sun can degrade epoxy resins. This is easily protected by a surface finish such as a coat of paint. Another factor is erosion or pitting caused by high speed impact with rain or dust particles. This is likely to occur on unprotected leading edges. There are surface finishes such as rain erosion coats and paints for preventing surface wear. Lightning strike is also a concern to composites. A direct strike can cause considerable damage to a laminate. Lightning strike protection in the form of conductive surfaces is applied in susceptible areas. In cases where substructure is also composite, the inside end of attachment bolts may need to be connected with each other and to ground by a conducting wire.

12.2.6 Joints

12.2.6.1 Mechanically-fastened joints

Successful joint design relies on knowledge of potential failure modes. Failure modes depend on joint geometry and laminate lay-up for one given material. The type of fastener used can also influence the occurrence of a particular failure mode. Different materials will give different failure modes.

Net-section tensile/compressive failures occur when the bolt diameter is a sufficiently large fraction of the strip width. For most successful designs, this fraction (D/W) is about one-quarter or more for near-isotropic lay-ups in carbon/epoxy systems that have a D/E of one-third or less.

Shear-out and shear-out delamination failures occur because the bolt is too close to the edge of the laminate. Such a failure can be triggered when there is only a partial net-section tension or bearing failure. D/t ratios should be 0.75 to 1.25.

In some instances the bolt head may be pulled through the laminate after the bolt is bent and deformed. This mode is frequently seen with countersunk fasteners and is highly dependent on the particular fastener used.

Bearing strength is a function of joint geometry, fastener and member stiffnesses. For a 0/±45/90 family of laminates with 20-40% of 0° plies and 40-60% of ±45° plies, plus a minimum (10%) of 90° plies, the bearing strength is relatively constant. Fastener characteristics such as clamp-up force and head configuration have a significant effect. However, for a specific laminate family, a specific fastener, and equal thickness laminate joining members, the parameter with the greatest influence is D/t.

Composite joints require smaller D/W and D/E ratios than do metals to get bearing failures.

Composite joint strength characteristics differ from metals because the strength is influenced by the bypass load going around the joint. This occurs when two or more fasteners are arranged in a line to transfer the load through a joint. Since not all of the load is reacted by one fastener, some of the load by-passes it. The by-pass effects become prominent once the ratio of by-pass to fastener bearing load exceeds 20%.

Titanium fasteners are the most common means of mechanical attachment in composites. This is because titanium is non-corrosive in the galvanic atmosphere created by the dissimilar materials. Titanium is closer to carbon on the cathodic scale.

12.2.6.2 Problems associated with adhesive bonding to peel-ply composite surfaces

There are two schools of thought in regard to the adhesive bonding of fibrous composite laminates. One demands light but thorough mechanical abrasion, such as by low-pressure grit blasting, because the only such bonds never to fail prematurely were made to abraded surfaces on completely dry laminates. The other permits bonding directly to surfaces created by stripping off peel plies, with or without a drying requirement, using the justification that there is “adequate” initial strength, even though some of these joints have failed prematurely in service. It is also significant that no ultrasonic inspection technique has been able to distinguish between bonded joints which will fail in service and those which will not. In addition, most traveler specimens do not represent the same cure conditions as experienced by adjacent large parts and, therefore, mechanical testing also often fails to identify defective bonding. One must depend on process control of techniques which can be relied upon 100 percent of the time and on thorough validation of the processes before committing them to production.

Consider a surface, created by stripping off a peel ply, which is then bonded as part of an adhesive joint. The resulting adhesive bond "sticks" well enough to pass all inspections; however, may fail prematurely at the interface between the laminate and the adhesive. All premature bond failures, other than those caused by incomplete cure, occur at the interface between the adhesive and the resin in the laminate. Structurally sound bonds either fail outside the joint area, cohesively within the layer of adhesive, or interlaminarly in the resin matrix between the surface fibers and the adhesive layer. These premature failures can occur either when uncured adhesive is bonded to precured laminates, or when uncured prepreg is cured against cured adhesive films used to stabilize honeycomb cores and the like.

There are several ways in which peel plies can create surfaces on which reliable durable bonds are not possible.

- The peel ply can be coated with a release agent, which transfers to the cured laminate when the peel ply is stripped off.

- The surface of the peel ply's fibers must be sufficiently inert that the ply can be removed without damaging the laminate. The grooves left in the laminate (or glue layer) by stripping off the peel ply may retain enough inert surface that the resin which is subsequently cured onto it may simply fail to adhere. Adhesion requires more than cleanliness; surface tension is also critical. In the absence of cohesion at the interface, a bonded joint relies only on mechanical interlocking, which is far weaker in peel than it is in shear.

- The peel-ply surface in the laminate consists of innumerable short grooves separated by sharp edges where the resin between the filaments in the peel ply fractured as the peel ply was stripped off. Moisture on (or in) the adhesive or the laminate can be trapped in these grooves. If this moisture cannot escape during the curing of the adhesive (or of a co-cured face sheet), the trapped moisture will result in a slick bond when examined microscopically after failure.

It should be noted that the latter two mechanisms function without any contamination.

One aircraft company's process specification has, for decades, required that any peel-ply surface to be bonded must first be thoroughly abraded to remove all traces of the texture of the peel ply. In the absence of the ridges between the grooves, it is presumed that moisture could escape, as it turned to steam during cure, unless the part was too large and too poorly ventilated. Using these requirements, this aircraft company has had no disbonds in those secondarily bonded composite structures which were grit blasted before bonding. The same cannot be claimed for bonds made to unabraded (or only scuff-sanded) peel-ply surfaces. In two instances, on different aircraft types, disbonds were traced to transfer of release agents from silicone-coated peel plies, the use of which is now banned throughout all documents, not only the approved materials lists.

On another aircraft type, interfacial failures on peel-ply surfaces appear to be the result of prebond moisture, the exact origin of which has yet to be established. An accident with one test panel during process qualification by a supplier revealed the consequences of condensate on adhesive film (the roll of adhesive had not been properly sealed when returned to the freezer after the previous use). There was absolutely no adhesion between the resin and the adhesive, even though the lap-shear numbers seemed to be acceptable. Microscopic examination of the surfaces clearly showed perfect imprints of the peel ply texture on both surfaces, with the surface in all grooves as smooth as glass and all of the resin on one surface and all of the adhesive on the other. However, with thicker-than-normal (0.123 inch (3.2 mm)) adherends of the same unidirectional carbon/epoxy, bonds made with the same nonreleased peel ply and the same kind of adhesive achieved cohesive failure of the bond at the same strength level attained by metal-to-metal bonding (6,000 psi (40 MPa) or so). With normal thickness composite adherends, only half this strength was reached, because the resin between the surface fibers and adhesive layer then failed in peel, leaving resin clearly covering both surfaces. This problem can be minimized by maintaining very tight time limits between making parts and bonding them together, with a requirement to thoroughly dry everything before bonding if the time constraints are exceeded. Careful scheduling can avoid this added drying step. The same high-strength cohesive bond failures had previously been achieved by another supplier of composite structures using grit-blasted surfaces and 0.080 inch (2.0 mm) thick unidirectional laminates.

In considering adhesive bond strength, it is vital to note that the specimen testing *validates the process, NOT the part*. There is no requirement for the specimen to look like the actual part. Indeed, in a properly designed bonded joint, the bond will not fail first. Consequently, the use of specimens which are “similar” to the part and which are evaluated in terms of the “adequacy” of the load carried in relation to the stresses in the part, is not sufficient to ensure the integrity of the bonded composite structure. This issue is complicated because, only with unidirectional tape laminates is it possible to develop sufficient load to fail a high-strength adhesive bond cohesively. Therefore, only such specimens can provide any assurance that the part they are intended to substantiate has been bonded properly. However, in real parts made from woven-fabric laminates, failures within bundles of fibers at 90° to the applied load will trigger interlaminate failures before such bond strengths can be attained.

In all cases, the one condition which can be detected visually on test specimens and failed parts alike which is a guaranteed indicator of a defective bond is an interfacial failure with all of the resin on one side and all of the adhesive on the other, with a clear imprint of the peel ply texture on both surfaces.

12.2.7 Design

The design of composite structure is complicated by the fact that every ply must be defined. Drawings or design packages must describe the ply orientation, its position within the stack, and its boundaries. This is straightforward for a simple, constant thickness laminate. For complex parts with tapered thicknesses and ply build-ups around joints and cutouts, this can become extremely complex. The need to maintain relative balance and symmetry throughout the structure increases the difficulty.

Composites can not be designed without concurrence. Design details depend on tooling and processing as does assembly and inspection. Parts and processes are so interdependent it could be disastrous to attempt sequential design and manufacturing phasing.

Another factor approached differently in composite design is the accommodation of thickness tolerances at interfaces. If a composite part must fit into a space between two other parts or between a substructure and an outer mold line, the thickness requires special tolerances. The composite part thickness is controlled by the number of plies and the per-ply-thickness. Each ply has a range of possible thicknesses. When these are layed up to form the laminate they may not match the space available for assembly within other constraints. This discrepancy can be handled by using shims or by adding "sacrificial" plies to the laminate (for subsequent machining to a closer tolerance than is possible with nominal per-ply-thickness variations). The use of shims has design implications regarding load eccentricities. Another approach is to use closed die molding at the fit-up edges to mold to exact thickness needed.

The anisotropy of special laminates, while more complicated, enables a designer to tailor a structure for desired deflection characteristics. This has been applied to some extent for aeroelastic tailoring of wing skins.

Composites are most efficient when used in large, relatively uninterrupted structures. The cost is also related to the number of detail parts and the number of fasteners required. These two factors drive designs towards integration of features into large cocured structures. The nature of composites enables this possibility. Well designed, high quality tooling will reduce manufacturing and inspection cost and rejection rate and result in high quality parts.

12.2.8 Handling and storage

Epoxy resins are the most common form of matrix material used in composites. Epoxies are perishable. They must be stored below freezing temperature and even then have limited shelf life. Once the material is brought out of storage there is limited time it can be used to make parts (30 days is common). For very complex parts with many plies, the material's permissible out-time can be a controlling factor. If the material is not completely used, it may be returned to storage. An out-time record should be kept. In addition, freezer storage of these materials is usually limited by the vendor to 6 to 12 months. Overage material will produce laminates with a high level of porosity.

The perishability of the material also requires that it be shipped refrigerated from the supplier. Upon arrival at the contractor's facility, there must be provisions to prevent it being left on-dock for long periods of time.

Tack is another composite material characteristic that is unique. Tack is "stickiness" of the prepreg. It is both an aid and a hindrance. Tack is helpful to maintain location of a ply once it is placed in position. It also makes it difficult to adjust the location once the ply has been placed.

12.2.9 Processing and fabrication

Composite parts are fabricated by successive placement of plies one after the other. Parts are built-up rather than machined down. Many metal fabrication steps require successive removal of material starting from large ingots, plates, or forgings. Prepreg "tape" material typically comes in rolls of relatively thin strips (0.005-0.015 inches or 0.13 - 0.38 mm). These strips are a variety of widths: 3", 6", and 36". Prepreg "fabric" is usually thicker than tape (0.007-0.020 inches or 0.18 - 0.51 mm) and usually comes in 36-inch (0.9 m) wide rolls.

Fabrication of a detail part requires the material to be taken out of the freezer in a sealed bag and allowed to come to room temperature prior to any operations. Placement of the prepreg on the tool (if not automated) requires care. The plies must be aligned properly to the desired angle and stacked in the prescribed sequence. Prepreg plies come with a backing material to keep them from sticking together on the rolls. This backing material must be removed to prevent contamination of the laminate. Care must be exercised when handling the material to prevent splinters from piercing the hands.

Part lay-up (particularly when done by hand) can lead to air entrapment between plies. This creates difficulty when the part is cured because the air may not escape, causing porosity. Thus, thick parts are normally pre-compacted using a vacuum periodically during the lay-up.

Some prepreg materials contain an excess of resin. This excess is expected to be "bled" away during cure. Bleeder plies are placed under the vacuum bag to soak up the excess resin. However, most current prepreg materials are "net resin" so no bleeding is required.

Composite processing requires careful attention to tool design. The tools must sustain high pressures under elevated temperature conditions. The composite material has different expansion characteristics than most tooling materials, thus thermal stresses are created in the part and in the tool. Tool surfaces are treated with a release agent to facilitate removal of the part after cure. Tools must also be pressure tight because autoclave processing requires application of a vacuum on the laminate as well as positive autoclave pressure. Lastly, tool design must account for the rate of manufacture and the number of parts to be processed.

Prepreg material is not fully cured. Curing requires application of heat and pressure that is usually performed in the autoclave. Autoclaves typically apply 85 psi (590 kPa) pressure up to 350°F (180°C). They can go beyond these values if required for other materials (such as polyimides), but they must be qualified for higher extremes. Autoclave size may limit the size of a part to be designed and manufactured. Very large autoclaves are available, but they are expensive and costly to run. Common problems that occur in autoclave operations include blown vacuum bags, improper heat-up rates, and loss of pressure.

Once the part is cured it may still require drilling, trimming and machining. Drilling of composites requires very sharp bits, careful feed and speed, and support of the back face to prevent splintering. Water-jet cutters are very useful for trimming. Machining produces a fine dust that requires protection for the operator's safety.

12.2.9.1 Quality control

The quality control function for composite materials starts at a much earlier phase than for metals. There is much coordination and interaction occurring between the material supplier and the user before the material is ever shipped. These controls are defined by the material and process specifications and in some cases design allowables requirements. The supplier is often required to perform chemical and mechanical tests on the material prior to shipment. These involve the individual material constituents, the prepreg, and cured laminates.

Material processing and handling must be monitored throughout the various manufacturing phases. Receiving inspections are performed on the prepreg and cured laminates when the material first comes in. From this time on the material is tracked to account for its shelf life and out-time.

Quality control activities include verification of the ply lay-up angle, its position in the stack, the number of plies, and the proper trim. During lay-up it is necessary to ensure all potential contaminates and foreign materials are not allowed to invade the material.

The curing process is monitored to ensure proper conformance to time-temperature-pressure profiles. These records are maintained for complete traceability of the parts.

After the part is cured, there are a number of methods to verify its adequacy. One of the most common is Through-Transmission-Ultrasonics (TTU). Parts with high porosity or delaminations can not transmit sound as well as unflawed parts. Thus ultrasound transmission is attenuated in a flawed part. Other techniques used to verify part quality include traveler specimens, specimens cut from excess material on the part, tracer yarns within the laminate, and in some cases proof loading. Visual inspections, thickness measurements, and tap testing also serve to interrogate composite parts.

One of the most crucial aspects of quality control is information on the effect of defects. It is not enough to discover a flaw or suspected non-conformity. There must also be sufficient information to evaluate the impact of that rejection. The quality control function in its entirety includes the dispositioning of exposed non-conformances. Dispositioning includes acceptance as-is, repair or rework, and scrappage. If proper dispositioning is not possible because of a lack of knowledge about the effect of defects, an inordinate expense will be incurred scrapping or reworking affected parts.

12.3 LESSONS LEARNED

12.3.1 Design and analysis

LESSON	REASON OR CONSEQUENCE
A-1. "Concurrent Engineering", whereby a new product or system is developed jointly and concurrently by a team composed of designers, stress analysts, materials and processes, manufacturing, quality control, and support engineers, (reliability, maintainability, survivability), as well as cost estimators, has become the accepted design approach.	To improve the quality and performance and reduce the development and production costs of complex systems
A-2. In general, design large cocured assemblies. Large assemblies must include consideration for handling and repair.	Lower cost due to reduced part count and assembly time. If the assembly requires overly complex tooling, the potential cost savings can be negated.
A-3. Structural designs and the associated tooling should be able to accommodate design changes associated with the inevitable increases in design loads.	To avoid scab-on reinforcements and similar last minute disruptions.
A-4. Not all parts are suited to composite construction. Material selection should be based on a thorough analysis that includes consideration of performance, cost, schedule, and risk.	The type of material greatly influences performance characteristics as well as producibility factors.
A-5. Uniwoven and bi-directional woven fabric should be used only when justified by trade studies (reduced fabrication costs). If justified, woven fabric may be used for 45° or 0°/90° plies.	Fabric has reduced strength and stiffness properties and the prepreg material costs more than tape. Fabric may be necessary for complex shapes and some applications may require the use of fabric for its drapeability.
A-6. Whenever possible, mating surfaces should be tool surfaces to help maintain dimensional control. If this is not possible, either liquid shims or, if the gap is large, a combination of precured and liquid shims should be used.	To avoid excessive out-of-plane loads that can be imposed if adjoining surfaces are forced into place. Large gaps may require testing.
A-7. Part thickness tolerance varies directly with part thickness; thick parts require larger tolerance.	Thickness tolerance is a function of the number of plies and the associated per-ply-thickness variation.
A-8. Carbon fibers must be isolated from aluminum or steel by using an adhesive layer and/or a thin glass-fiber ply at faying surfaces.	Galvanic interaction between carbon and aluminum or steel will cause corrosion of the metal.

A-9. The inspectability of structures, both during production and in-service, must be considered in the design. Large defects or damage sizes must be assumed to exist when designing composite structures if reliable inspection procedures are not available.	There is a much better chance that problems will be found if a structure is easily inspected.
A-10. In Finite Element Analysis (FEA) a fine mesh must be used in regions of high stress gradients, such as around cut-outs and at ply and stiffener drop-offs.	Improper definition or management of the stresses around discontinuities can cause premature failures.
A-11. Eliminate or reduce stress risers whenever possible.	Composite (fiber-dominated) laminates are generally linear to failure. The material will not yield locally and redistribute stresses. Thus, stress risers reduce the static strength of the laminate.
A-12. Avoid or minimize conditions which cause peel stresses such as excessive abrupt laminate terminations or cocured structures with significantly different flexural stiffnesses (i.e., $EI_1 >> EI_2$).	Peel stresses are out-of-plane to the laminate and hence, in its weakest direction.
A-13. Buckling or wrinkling is permissible in thin composite laminates provided all other potential failure modes are properly accounted for. In general, avoid instability in thick laminates.	Significant weight savings are possible with postbuckled design.
A-14. Locating 90° and ±45° plies toward the exterior surfaces improves the buckling allowables in many cases. Locate 45° plies toward the exterior surface of the laminate where local buckling is critical.	Increases the load carrying capability of the structure.
A-15. When adding plies, maintain balance and symmetry. Add between continuous plies in the same direction. Exterior surface plies should be continuous.	Minimizes warping and interlaminar shear. Develops strength of plies. Continuous surface plies minimize damage to edge of ply and help to prevent delamination.
A-16. Never terminate plies in fastener patterns.	Reduces profiling requirements on substructure. Prevents delamination caused by hold drilling. Improves bearing strength.
A-17. Stacking order of plies should be balanced and symmetrical about the laminate midplane. Any unavoidable unsymmetric or unbalanced plies should be placed near the laminate midplane.	Prevents warpage after cure. Reduces residual stresses. Eliminates "coupling" stresses.

A-18. Use fiber dominated laminate wherever possible. The [0°/±45°/90°] orientation is recommended for major load carrying structures. A minimum of 10% of the fibers should be oriented in each direction.	Fibers carry the load; the resin is relatively weak. This will minimize matrix and stiffness degradation.
A-19. When there are multiple load conditions, do not optimize the laminate for only the most severe load case.	Optimizing for a single load case can produce excessive resin or matrix stresses for the other load cases.
A-20. If the structure is mechanically fastened, an excess of 40% of the fibers oriented in any one direction is inadvisable.	Bearing strength of laminate is adversely affected.
A-21. Whenever possible maintain a dispersed stacking sequence and avoid grouping similar plies. If plies must be grouped, avoid grouping more than 4 plies of the same orientation together.	Increases strength and minimizes the tendency to delaminate. Creates a more homogeneous laminate. Minimizes interlaminar stresses. Minimizes matrix microcracking during and after service.
A-22. If possible, avoid grouping 90° plies. Separate 90° plies by a 0° or ±45° plies where 0° is direction of critical load.	Minimizes interlaminar shear and normal stresses. Minimizes multiple transverse fracture. Minimizes grouping of matrix critical plies.
A-23. Two conflicting requirements are involved in the pairing or separating of ±θ° plies (such as ±45°) in a laminate. Laminate architecture should minimize interlaminar shear between plies and reduce bending/twisting coupling.	Separating ±θ° plies reduces interlaminar shear stresses between plies. Grouping ±θ° plies together in the laminate reduces bending/twisting coupling.
A-24. Locate at least one pair of ±45° plies at each laminate surface. A single ply of fabric will suffice.	Minimizes splintering when drilling. Protects basic load carrying plies.
A-25. Avoid abrupt ply terminations. Try not to exceed dropping more than 2 plies per increment. The plies that are dropped should not be adjacent to each other in the laminate.	Ply drops create stress concentrations and load path eccentricities. Thickness transitions can cause wrinkling of fibers and possible delaminations under load. Dropping non-adjacent plies minimizes the joggle of other plies.
A-26. Ply drop-offs should not exceed 0.010 inch (0.25mm) thick per drop with a minimum spacing of 0.20 inch (0.51 mm) in the major load direction. If possible, ply drop-offs should be symmetric about the laminate midplane with the shortest length ply nearest the exterior faces of the laminate. Shop tolerance for drop-offs should be 0.04 inch (1 mm).	Minimizes load introduction into the ply drop-off creating interlaminar shear stresses. Promotes a smooth contour. Minimizes stress concentration.

A-27. Skin ply drop-offs should not occur across the width of spars, rib, or frame flange.	Provides a better load path and fit-up between parts.
A-28. In areas of load introduction there should be equal numbers of +45° and -45° plies on each side of the mid-plane.	Balanced and symmetric pairs of ±45° plies are strongest for in-plane shear loads which are common at load introduction points.
A-29. A continuous ply should not be butt-spliced transverse to the load direction.	Introduces a weak spot in the load path.
A-30. A continuous ply may be butt-spliced parallel to the load direction if coincident splices are separated by at least four plies of any orientation.	Eliminates the possibility of a weak spot where plies are butted together.
A-31. The butt joint of plies of the same orientation separated by less than four plies of any direction must be staggered by at least 0.6 inch (15 mm).	Minimizes the weak spot where plies are butted together.
A-32. Overlaps of plies are not permitted. Gaps should not exceed 0.08 inch (2 mm).	Plies will bridge a gap, but must joggle over an overlap.

12.3.1.1 Sandwich design

B-1. Facesheets should be designed to minimize people induced damage during handling or maintenance of component.	Thin skin honeycomb structure is very susceptible to damage by harsh handling.
B-2. When possible avoid laminate buildup on the core side of the laminate.	Minimizes machining of the core.
B-3. Core edge chamfers should not exceed 20° (from the horizontal plane). Larger angles may require core stabilization. Flex core is more sensitive than rigid core.	Prevents core collapse during cure cycle.
B-4. Use only non-metallic or corrosion resistant metal honeycomb core in composite sandwich assemblies.	Prevents core corrosion
B-5. Choice of honeycomb core density should satisfy strength requirements for resisting the curing temperature and pressure during bonding or cocuring involving the core. 3.1 PCF (50 g/m^3) is a minimum for non-walking surfaces.	Prevents crushing of the core.

B-6. For sandwich structure used as a walking surface, a core density of 6.1 PCF (98 g/m^3) is recommended.

3.1 PCF (50 g/m^3) core density will result in heel damage to the walking surface.

B-7. Do not use honeycomb core cell size greater than 3/16 inch (4.8 mm) for cocuring sandwich assemblies (1/8 inch (3.2 mm) cell size preferred).

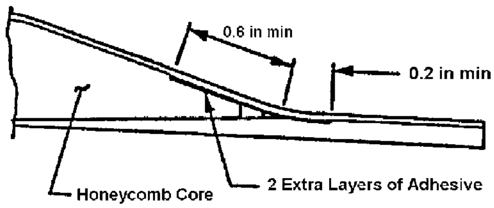
Prevents dimpling of face sheets.

B-8. When core is required to be filled around bolt holes, etc., this should be done using an approved filler to a minimum of 2D from the bolt center.

Prevents core crushing and possible laminate damage when bolt is installed.

B-9. Two extra layers of adhesive should be applied to the inner moldline at the core run out (edge chamfer). This should be applied a minimum of 0.6 in. (15 mm) from the intersection of the inner skin and edge band up the ramp and a minimum of 0.2 in. (5 mm) from that point into the edge band.

Curing pressures tend to cause the inner skin to "bridge" in this area creating a void in the adhesive (skin to core bond).



B-10. The use of honeycomb sandwich construction must be carefully evaluated in terms of its intended use, environment, inspectability, repairability, and customer acceptance.

Thin skin honeycomb is susceptible to impact damage, water intrusion due to freeze/thaw cycles, and is difficult to repair.

12.3.1.2 Bolted joints

C-1. Design the joints first and fill in the basic structure afterwards.

Optimizing the “basic” structure first compromises the joint design and results in low overall structural efficiency.

C-2. Joint analysis should include the effects of shimming to the limits permitted by drawings.

Shimming can reduce joint strength.

C-3. Design joints to accommodate the next larger fastener size.

To accommodate routine MRB and repair activities.

C-4. Bolted joint strength varies far less with percentage of 0° plies in fiber pattern than does unnotched laminate strength.

The stress concentration factor, K_t, is highly dependent on 0° plies.

C-5. Optimum single-row joints have approximately three-fourths of the strength of optimum four-row joints.

Optimum single-row joints operate at higher bearing stress than the most critical row in an optimized multi-row joint.

C-6. Common errors in composite bolted joints are to use too few bolts, space them too far apart, and to use too small a diameter.	Does not maximize the strength of the laminate.
C-7. Rated shear strength of fasteners does not usually control the joint design.	Bolt diameter is usually governed by the need not to exceed the allowable bearing stress in the laminate.
C-8. The peak hoop tensile stress around bolt holes is roughly equal to the average bearing stress.	Keeping the laminate tensile strength high requires keeping the bearing stress low.
C-9. Maximum torque values should be controlled, particularly with large diameter fasteners.	Avoids crushing the composite.
C-10. Bolt bending is much more significant in composites than for metals.	Composites tend to be thicker (for a given load) and more sensitive to non-uniform bearing stresses (because of brittle failure modes.
C-11. Optimum w/d ratio for multi-row bolted joints varies along length of joint. w/d = 5 at first row to minimize load transfer, w/d = 3 at last row to maximize transfer, w/d = 4 for intermediate bolts.	Maximizes joint strength.
C-12. Stainless steel fasteners in contact with carbon should be permanent and installed wet with sealant.	Prevents galvanic corrosion.
C-13. Use a layer of fiberglass or Kevlar (0.005 inch (0.13 mm) minimum) or adhesive with serim on faying surfaces of carbon epoxy panels to aluminum.	Prevents corrosion of aluminum.
C-14. Bolt stresses need careful analysis, particularly for the effects of permissible manufacturing parameters, for example, hole perpendicularity (±10°), shimming, loose holes.	Bolt failures are increasingly becoming the “weak link” with current high strength composite materials.
C-15. Bolted joint data bases should include the full range of all permitted design features.	Establishes that failure modes remain consistent and that there are no detrimental interaction effects between design parameters.
C-16. The design data base should be sufficient to validate all analysis methods over the entire range permitted in design.	For proper verification of analytical accuracy.

C-17. Mechanical joint data bases should contain information pertaining to durability issues such as clamp-up, wear at interfaces, and hole elongation. Manufacturing permitted anomalies such as hole quality, edge finish, and fiber breakout also need to be evaluated.	Practical occurrences can affect strength and durability.
C-18. Use drilling procedures that prevent fiber break out on the back side of the component.	Improper back side support or drilling procedures can damage surface plies on the back side.
C-19. Splice plate stresses should be lower than the stresses in skins to prevent delaminations.	Splice plates see less clamp up than the skin sandwiched in between, because of bolt bending.
C-20. The best bolted joints can barely exceed half the strength of unnotched laminates.	The strength reduction is caused by stress concentrations around the hole for the fastener.
C-21. Laminate percentages for efficient load transfer: 0° = 30-50%; $\pm 45^\circ$ = 40-60%; 90° = minimum of 10%.	Best range for bearing and by-pass strength.
C-22. Countersink depths should not exceed 70% of laminate thickness.	Deep countersinks result in degraded bearing properties and increased hole wear.
C-23. Fastener edge distance and pitch: Use 3.0D edge distance in direction of major load; use 2.5D + 0.06 side distance. (D is diameter of fastener.)	Maximizes joint strength.
C-24. Gap between attached parts should not exceed 0.03 inch (0.8 mm) for non-structural shim.	Large gaps cause excessive bolt bending, non-uniform bearing stresses, and eccentric load path.
C-25. Any gap in excess of 0.005 inch should be shimmed.	Minimizes interlaminar stresses due to clamp-up.
C-26. Use “form-in-place” gaskets on carbon/epoxy doors over anodized aluminum substructure. Allow for a seal thickness of 0.010 ± 0.005 inch (-0.25 ± 0.13 mm) minimum.	Prevents corrosion of aluminum.
C-27. Use only titanium, A286, PH13-8 MO, monel or PH17-4 stainless steel fastener with carbon/epoxy.	Prevents galvanic corrosion.
C-28. Do not buck rivets in composite structure.	The bucking force can damage the laminate.

C-29. The use of interference fit fasteners should be checked before permitting their use in design.	Installation of interference-fit fasteners can damage laminates if a loose-fit sleeve is not installed first.
C-30. Fastener-to-hole size tolerance for primary structure joints must be assessed and controlled.	Tight fitting fastener promotes uniform bearing stress in a single fastener hole, and promotes proper load sharing in a multi-fastener joint.
C-31. Squeeze rivets can be used if washer is provided on tail side.	Washer helps protect the hole.
C-32. For blind attachments to composite substructure, use fastener with large blind side footprint of titanium or A286.	Prevents damage to composite substructure by locking collars of fasteners.
C-33. Tension head fasteners are preferred for most applications. Shear head fasteners may be used in special applications only with stress approval.	Shear head fasteners.
C-34. Avoid putting fastener threads in bearing against the laminate.	Fastener threads can gouge and damage the laminate.
C-35. Tapered splice plates should be used to tailor the load transfer, row by row, to minimize the bearing stress at the most critical row.	Multi-row bolted joints between uniformly thick members will have high peak bearing loads in outermost rows of fasteners.

12.3.1.3 Bonded joints

D-1. Use secondary adhesive bonding extensively for thin, lightly loaded, composite structures, restricting the use of mechanical fastening to thicker, more heavily loaded structures.	Reduces cost. Reduces the number of holes in composite components. Reduces weight by eliminating build-ups for fastener countersinking and bearing strength.
D-2. Never design for an adhesive bond to be the weak link in a structure. The bonds should always be stronger than the members being joined.	Maximizes the strength of the structure. The bond could act as a weak-link fuse and unzip catastrophically from a local defect.
D-3. Thick bonded structures need complex stepped-lap joints to develop adequate efficiency.	Large loads require many steps to transfer the load and assure that adhesive develops the strength of the adherends.
D-4. Anticipate bolted repairs for thick structures by reducing strain levels.	Thick structures are impractical to repair by bonding, except for one-shot and throwaway structures.

D-5.	When there is no need for repair, as in missiles and unmanned aircraft, bonding permits extremely high structural efficiencies to be obtained, even on thick structures.	Load transfer is performed without drilling holes for fasteners.
D-6.	Proper surface preparation is a "must" - beware of "cleaning" solvents and peel plies. Mechanical abrasion is more reliable.	Maintaining joint strength in service is very dependent on the condition of the surfaces to be bonded.
D-7.	Laminates must be dried before performing bonded repairs.	Heat applied to the laminates during repair can cause any moisture present to vaporize and cause blisters.
D-8.	Adherend overlaps must not go below specified minimums.	Key to durability of bonded joints is that some of the adhesive must be lightly stressed to resist creep.
D-9.	Bonded overlaps are usually sized to survive hot/wet environmental conditions.	Elevated temperature and moisture degrade the strength and stiffness of the adhesive.
D-10.	Bonded joint strength can also be degraded by cold environment where adhesive is brittle.	The brittleness of the adhesive limits joint strength.
D-11.	Taper ends of bonded overlaps down to 0.020 inch (0.51 mm) thick with a 1-in-10 slope.	Minimizes induced peel stresses that would cause premature failures.
D-12.	Adhesives work best in shear, are poor in peel, but composites are even weaker in interlaminar tension.	Joint must be designed to minimize out-of-plane stresses.
D-13.	Design of simple, uniformly thick (for near quasi-isotropic carbon/epoxy) bonded splices is very simple. Use 30 t overlap in double shear, 80 t overlap for single-lap joints, 1-in-50 slope for scarf joint.	Provides a bonded joint with good strength capability.
D-14.	Design of stepped-lap joints for thick structure needs a nonlinear analysis program.	Complex stress states in stepped-lap joints. Nonlinear adhesive characteristics.
D-15.	Adhesives are well characterized by thick-adherend test specimen, generating complete nonlinear shear stress-strain curve.	This test provides ample data for analysis of joints critical in shear.
D-16.	For highly loaded bonded joints a co-cured, multiple step, double sided lap is preferred.	Very efficient joint design.

D-17. Never design a bonded joint such that the adhesive is primarily loaded in either peel or cleavage.	Adhesive peel strength is very poor and unpredictable.
D-18. Ductile adhesives are preferred over brittle ones.	Ductile adhesives are more forgiving.
D-19. Film adhesives are preferred over paste adhesives for large area bonds.	Provides more uniform bond line, easier to contain when heated.
D-20. Balanced adherend stiffnesses improve joint strength.	Reduces peel stresses.
D-21. Minimize joint eccentricities.	Reduces peel stresses.
D-22. Use adherends of similar coefficients of thermal expansion.	Reduces residual stresses.
D-23. Insure the bonded joint configuration is 100% visually inspectable.	Improves reliability and confidence. Need to emphasize process control.

12.3.1.4 Composite to metal splice joints

E-1. Bonding composites to titanium is preferred; steel is acceptable; aluminum is not recommended.	Minimizes differences in thermal expansion coefficient.
E-2. Bonded step joints preferred over scarf joints.	Better fit, higher strength.
E-3. Where possible, 45° plies (primary load direction) should be placed adjacent to the bondline; 0° plies are also acceptable. 90° plies should never be placed adjacent to the bondline unless it is also the primary load direction.	Minimizes the distance between the bondline and the plies that carry the load. Prevents failure of surface ply by “rolling log" mechanism.
E-4. For a stepped joint, the metal thickness at the end step should be 0.030 inch (0.76 mm) minimum and the step no longer than 0.375 in (9.5 mm).	Prevents metal failure of end step.
E-5. If possible, have ±45° plies end on first and last step of bonded step joint.	Reduces peak interlaminar shear stresses at end steps.
E-6. If possible, do not end more than two 0° plies (not more than 0.014 inch (0.36 mm) maximum thickness) on any one step surface. For 0° plies ending on last step (longest 0° ply) serrated edges have been shown to reduce stress concentration.	Reduces stress concentration at end of joint.

E-7. or 90° plies should butt up against the first step of a step joint.	Reduces magnitude of interruption in load path.
E-8. Tension and peel stresses should be avoided in adhesive bonded joints.	Minimum strength direction of adhesive.

12.3.1.5 Composite to metal continuous joints

F-1. Bonding composites to titanium preferred, steel is acceptable.	Minimizes differences in thermal expansion coefficient.
F-2. No composite to aluminum structural adhesive bond except for corrosion resistant aluminum honeycomb core and lightly loaded secondary structure.	Minimizes interlaminar shear stress due to large difference in thermal expansion coefficient between composites and aluminum.

12.3.1.6 Composite to composite splice joints

G-1. Scarfed joints are never preferred over stepped joints, except for repairs of thin structures.	Improves strength of joint.
G-2. Cocured joints are preferred over precured joints if there are fit-up problems.	Less sensitive to tolerance mismatches.
G-3. For pre-cured parts, machined scarfs are preferred over layed up scarfs.	For improved fit.
G-4. Use of cocured bonded subassemblies should be evaluated in terms of supportability.	Reduces ply count and assembly time, but increases rework cost.
G-5. Bonded repairs are not acceptable for thick laminates.	Taper ratio requirement makes bonded repair impractical.

12.3.2 Materials and processes

LESSON	**REASON OR CONSEQUENCE**
H-1. Materials selection forms the foundation for structural and manufacturing development and supportability procedures.	The material selected influences critical issues, how parts are fabricated, inspected, and assembled, and how much previous data/learning is available.
H-2. Material selection must be based on a thorough analysis and occur early in the process.	Various materials have various advantages. Specific applications should use materials that best fit the needs of the application.

H-3. Imide-based polymer composites should consider galvanic degradation.	Some of these materials have exhibited galvanic corrosion in the presence of salt water.
H-4. Net-resin prepregs improve quality at reduced cost.	Minimizes (eliminates) bleeding of prepreg during cure.
H-5. Composite material applications must have a margin between the wet T_g and the use temperature (usually 50°F).	To prevent the material from operating in an environment where its properties become greatly decreased and widely scattered.
H-6. Specific issues impacting materials selection/use: Fluid/solvent degradation High residual thermal stresses Mechanical performance Out-time/tack time Effects of defects Sensitivity to processing variations OSHA/EPA requirements Cost (Procurement, Manufacturing, Quality) Environmental degradation Cocure compatibility with other composites and adhesives.	Ignoring key material features could result in an inferior product.

12.3.3 Fabrication and assembly

LESSON	REASON OR CONSEQUENCE
I-1. Highly integral cocured structures are weight and cost effective, however, they place a high burden on tooling design.	Integrally cured structure eliminates parts and fasteners. The tools to perform the fabrication are complex and greatly influence the quality of the part.
I-2. Machining/drilling must be rigorously controlled; this includes feeds, speeds, lubrication, and tool replacement.	Backside breakout is a major nonconformance on all programs. Composite to metal drilling must avoid chip scoring. Highly directionally stacked laminates tend to gouge during drilling in the stacked areas.
I-3. Waterjet trimming of cured laminates has been shown to be highly successful.	Produces a clean, smooth edge very rapidly.
I-4. Sanding/trimming must consider out-of-plane damage. Tool rotation must be in the same plane as the laminate.	These operations tend to produce forces in the weakest direction of the laminate.

I-5.	Waterjet prepreg cutting can fray prepreg edges. Frequent nozzle replacement may be necessary.	Produces acceptable cuts.
I-6.	Lay-up shop temperature/humidity directly impacts handleability.	Tack and drapeability are influenced by temperature and moisture in the prepreg.
I-7.	Unauthorized hand creams can lead to extensive porosity and contamination. The use of gloves can prevent this risk.	Some hand creams contain ingredients that are contaminates.
I-8.	Irons and hot air guns used for ply locating and compaction must be calibrated.	Avoids ply damage due to overheating.
I-9.	FOD control in the lay-up shop is absolutely necessary.	Can lead to foreign materials in the laminate.
I-10.	Hand drills can cause significant damage.	Feed and speed are less precise. Hole perpendicularity may be imperfect.
I-11.	Ply placement tolerances must be able to meet design requirements.	Strength/stiffness analysis is based on assumptions regarding angle of the plies and their location.
I-12.	Assembly jigs must provide the dimensional rigidity necessary to meet assembly tolerances.	Composites are less tolerant of pull-up stresses imposed by poor fit.
I-13.	Engineering drawings and specifications should be supplemented by fully illustrated planning documents or handbooks.	Drawings and specifications tend to be highly complex and detailed. They are not easy to follow on the factory floor.
I-14.	Consider two-step curing process in bonding and cocuring operations.	Alleviates problems such as core slippage and crushing, skin movement, and ply wrinkling.
I-15.	Fastener grip lengths should take into account actual thicknesses (including shims) at the fastener location.	A fastener with excessive grip length may not provide proper clamp-up. Too short a grip length may put threads in bearing or result in an improperly formed head.
I-16.	Tolerance requirements have a big impact on selecting manufacturing and tooling processes and therefore cost.	Different processes produce varying tolerance control.
I-17.	If possible, the mating surfaces should be tool surfaces.	Maintains the best possible dimensional control.

I-18. The use of molded rubber and trapped rubber tools has had mixed success. Rubber can be used successfully in local areas as a pressure intensifier, such as inside radii on stiffeners of cocured structure.	Rubber tools are difficult to remove, tend to become entrapped. They do not wear well.
I-19. Analyses can be done to predict distortion or "spring back" of a part after it is removed from a tool. The problem is usually solved by trial and error methods through tool modifications. The "spring back" problem is generally more pronounced on metal tools than on CFRP tools.	Residual or curing stresses build up in composite laminates formed to various shapes. When the structure is removed from the tool, the residual stresses tend to relieve themselves causing "spring back".
I-20. Tool design, including tool material selection, must be an integral part of the overall design process.	Tool design is dependent on part size and configuration, production rate and quantity, and company experience.
I-21. Aluminum tools have been used successfully on small parts but are avoided on large parts and female molds.	Thermal expansion mismatch.
I-22. Invar is often used for production tooling.	Invar has good durability and low thermal expansion.
I-23. Electroformed nickel also produces a durable, high quality tool, but is less frequently used.	More expensive.
I-24. Steel or Invar tools are needed for curing high temperature resins such as polyimides and bismaleimides.	The thermal mismatch with other materials is magnified at the higher cure temperature of these resins.
I-25. Air bubbles in a silicone rubber tool will cause "bumps" in the cured laminate.	The tool fails to provide support for the laminate and apply uniform pressure.
I-26. Resin containment is essential to part thickness control.	Uncontained resin will cause resin rich and resin starved areas.

12.3.4 Quality control

LESSON	REASON OR CONSEQUENCE
J-1. Continuing process control and process monitoring are required during production.	Assures that neither the process nor the material is changing.

J-2. Ultrasonic C-Scan is the most commonly used NDI technique. It may be supplemented by other techniques such as X-ray, shearography, and thermography.	Useful for detecting porosity, disbonds and delaminations.
J-3. Determine and understand the effect of defects on part performance.	Minimizes the cost of MRB activity.
J-4. There is no substitute for destructive, tear-down inspections of complex parts under development.	Not all discrepancies can be detected by NDI methods.

12.3.5 Testing

LESSON	REASON OR CONSEQUENCE
K-1. The testing of joints and demonstration of damage tolerance should include sufficient detail to adequately evaluate structural details and size effects.	Small details and size effects can have a large influence on the response of composite structure. In general, damage tolerance of composites exhibits size effects. Bolted and bonded joints, if properly designed, do not.
K-2. A well planned test program must include an accelerated approach for taking into account the effects of moisture, temperature, impact damage, etc.	Including moisture and elevated temperature on a real-time basis for full-scale testing is impractical for most components.
K-3. A finite element analysis should be performed prior to conducting a full-scale test. The analysis must accurately simulate the test article and the boundary conditions of the test fixture and loads applied during the test.	For a more accurate assessment of the internal loads and failure prediction of the test article.
K-4. Traceability of test specimens to batch, constituent material lots, autoclave run, panel, position in panel, and technicians is essential to data analysis.	If full traceability is not maintained and documented, the cause of outlier data points or unexpected failure modes may be difficult to identify. The result is that "bad" data, which might legitimately be discarded for cause, might be retained and add undeserved variability to the data set.
K-5. Adequate instrumentation is essential for all design/development or concept validation testing. Placement of strain gages, LVDT's, etc., should be based on analysis.	A good understanding of local failure modes and correlation of test results with analysis will aid the design process.

12.3.6 Certification

LESSON	REASON OR CONSEQUENCE
L-1. The "building block" approach is an excellent method for developing and validating the details of the design.	A wide variety of issues and details can be evaluated cost effectively. Hardware serves a dual purpose - engineering and manufacturing.
L-2. Component qualification is complicated by the fact that critical design conditions include hot, wet environments. This is often accomplished by overloading a test article that is in ambient conditions, or by analysis of failure modes coupled with strain measurements related back to subcomponent hot, wet tests.	It is generally impractical to try to ingest moisture in full scale test articles and test them hot.

12.3.7 In-service and repair

LESSON	REASON OR CONSEQUENCE
M-1. In spite of concerns about the sensitivity of composites to damage, experience in service has been good. Navy aircraft have not experienced any delamination failures in service. Most damage has occurred during assembly or routine service performed on the aircraft.	Current design, fabrication, and certification procedures adequately prepare the structure to survive its intended environment.
M-2. Composite components located in the vicinity of engine exhaust are subject to thermal damage. At present there are no acceptable NDI methods for detecting thermal damage of matrix materials.	Composite components exposed to engine exhaust or other heat sources should be shielded or insulated to keep temperatures down to an acceptable level.
M-3. Moisture ingestion is the biggest problem with honeycomb sandwich structure. The thin, stabilized skins that make honeycomb structurally efficient are also the reason they are damage prone. Panels get walked on and damaged.	Honeycomb design must be applied judiciously. Repair must account for the possibility of water in the core.
M-4. Aircraft are commonly painted and repainted. Paint stripping has been done with solvents. Solvents can damage epoxy matrices.	Increased use of water-based paints and solvent-less stripping of paint is desirable.

M-5. Records pertaining to MRB actions and in plant repairs of composite parts should be readily available to personnel responsible for in-service maintenance.	During routine maintenance checks, depot personnel sometimes find defects or discrepancies. In some cases they have been able to determine that the "defect" was in the part at delivery and considered acceptable.
M-6. Supportability and repair must be responsive to service environment.	It is necessary to account for equipment, facilities, and personnel capabilities.

REFERENCES

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