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DESIGN CRITERIA
(STRUCTURES)

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ADVANCED COMPOSITE STRUCTURES

DECEMBER 1974

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

NASA experience has indicated a need for uniform criteria for the design of space vehicles. Accordingly, criteria are being developed in the following areas of technology:

Environment
Structures
Guidance and Control
Chemical Propulsion

Individual components of this work will be issued as separate monographs as soon as they are completed. A list of all published monographs in this series can be found at the end of this document.

These monographs are to be regarded as *guides* to the formulation of design requirements and specifications by NASA Centers and project offices.

This monograph was prepared under the cognizance of the Langley Research Center. The Task Managers were W. C. Thornton and J. R. Hall. The authors were C. W. Rogers and D. L. Reed of General Dynamics Corporation. A number of other individuals assisted in developing the material and reviewing the drafts. In particular, the significant contributions made by the following are hereby acknowledged: H. P. Adam, F. Cherry, L. B. Greszczuk, and D. M. Purdy of McDonnell Douglas Corporation; F. J. Darmes of Rockwell International Corporation; J. R. Eisenmann, B. E. Kaminski, M. R. Scales, and D. J. Wilkins of General Dynamics Corporation; R. N. Hadcock of Grumman Aerospace Corporation; R. R. June of The Boeing Company; L. W. Lassiter of Lockheed-Georgia Company; and J. P. Peterson and C. C. Poe of NASA Langley Research Center.

NASA plans to update this monograph periodically as appropriate. Comments and recommended changes in the technical content are invited and should be forwarded to the attention of the Structures and Dynamics Division, Langley Research Center, Hampton, Virginia 23665.

December 1974

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GUIDE TO THE USE OF THIS MONOGRAPH

The purpose of this monograph is to provide a uniform basis for design of flightworthy structure. It summarizes for use in space vehicle development the significant experience and knowledge accumulated in research, development, and operational programs to date. It can be used to improve consistency in design, efficiency of the design effort, and confidence in the structure. All monographs in this series employ the same basic format – three major sections preceded by a brief INTRODUCTION, Section 1, and complemented by a list of REFERENCES.

The STATE OF THE ART, Section 2, reviews and assesses current design practices and identifies important aspects of the present state of technology. Selected references are cited to supply supporting information. This section serves as a survey of the subject that provides background material and prepares a proper technological base for the CRITERIA and RECOMMENDED PRACTICES.

The CRITERIA, Section 3, state *what* rules, guides, or limitations must be imposed to ensure flightworthiness. The criteria can serve as a checklist for guiding a design or assessing its adequacy.

The RECOMMENDED PRACTICES, Section 4, state *how* to satisfy the criteria. Whenever possible, the test procedure is described; when this cannot be done, appropriate references are suggested. These practices, in conjunction with the criteria, provide guidance to the formulation of requirements for vehicle design and evaluation.

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ADVANCED COMPOSITE STRUCTURES

1. INTRODUCTION

Advanced composite materials are defined as high-modulus, high-strength, continuous fibers of boron, graphite, or polymeric material embedded in a polymer or metal matrix. These composite materials are highly orthotropic and possess little or no ductility. Efficient use of composites requires tailoring of the strength and stiffness of the laminates to meet the local conditions. However, the current practice in designing with composites is to use design criteria that are the outgrowth of years of experience with such homogeneous materials as steel and aluminum, which have failure or fracture modes that are quite different from those of composite materials.

Further, although the use of metals-oriented design criteria is often the only feasible approach to the design of a composite structure, this procedure does not permit realization of the full potential of the composite materials. In fact, it may not even ensure structural integrity unless careful attention is given to the special problems associated with composite structure, such as scale effects and brittleness.

Inadequate attention to such problems can result in premature failure of composite structures, as the following examples indicate:

- A wing box failed at 60 percent of ultimate load because of stress concentration at a panel corner.
- A horizontal tail failed at 91 percent of ultimate load because of stress concentration in a bonded joint.
- A cryogenic tankage support strut failed at slightly over limit load [tension at 90K (-300°F)] because of a bondline deficiency.
- A horizontal tail failed at 79 percent of ultimate load because of a skin-to-main-spar bond failure or a tensile-type skin rupture.
- A horizontal tail failed at 74 percent of ultimate load because of stress concentration in the laminate resulting from an attachment fastener pattern that was not properly accounted for.
- A wing skin failed at 47 percent of ultimate load for the same reason.

This monograph establishes structural design criteria and recommends practices to ensure the design of sound composite structures, including composite-reinforced metal structures. It does not discuss design criteria for fiber-glass composites and such advanced composite materials as beryllium wire or sapphire whiskers in a matrix material. Although the criteria were developed for aircraft applications, they are general enough to be applicable to space vehicles and missiles as well.

The monograph covers four broad areas: (1) materials, (2) design, (3) fracture control, and (4) design verification. The materials portion deals with such subjects as material system design, material design levels, and material characterization. The design portion includes panel and joint design, applied loads, internal loads, design factors, reliability, and maintainability. Fracture control includes such items as stress concentrations, service-life philosophy, and the management plan for control of fracture-related aspects of structural design using composite materials. Design verification discusses ways to prove flightworthiness to the customer.

The main parameters that influence design criteria for composite structures are the physical and mechanical properties of composites. These properties are a result of the basic material form, which is a layer of parallel fibers embedded in a matrix material. The combination forms a lightweight, orthotropic lamina (a ply with high modulus and high strength in the fiber direction) which exhibits material properties quite different from those of conventional homogeneous metallic materials. These properties, in conjunction with the low coefficient of thermal expansion of some systems, make possible many applications for composites in components that require great dimensional stability, such as space antennas.

This monograph is related to other monographs in this series in several areas. Design practices, failure-mode analyses, and design data can be found in the monographs on circular cylinders (ref. 1) and structural plates (ref. 2). Proof-testing concepts for metallic pressure vessels, which are generally applicable to composite pressure vessels, are treated in the monographs on fracture control of metallic pressure vessels (ref. 3) and space shuttle structures (ref. 4). Applicable practices for preparing a test plan, determining the type of data required, and preparing the necessary documentation are given in the monographs on qualification testing (ref. 5) and acceptance testing (ref. 6). Vibration testing does not depend upon the structural material; thus, the practices contained in the monograph on vibration prediction (ref. 7) are applicable to composite structures. The analytical techniques in the monograph on discontinuity stresses in metallic pressure vessels (ref. 8) are also generally applicable to composite pressure vessels since most of the listed computer codes already accommodate composite materials.

2. STATE OF THE ART

Composite-materials technology came into being with the development of boron filaments in the early 1960's. The U.S. Air Force conducted studies in 1963 and 1964 that promised a weight savings of 25 to 50 percent from use of boron composite in aerospace structural applications. As a result of these studies, the Air Force Materials Laboratory at Wright-Patterson Air Force Base initiated an Advanced Development Program and then in 1965 formed the Advanced Composites Division. These efforts have been reinforced by related activities in industry and other governmental agencies.

The first hardware efforts using advanced composite materials were begun in 1965. Two of these programs involved building a reduced-scale section of the structural box of the horizontal stabilizer on the F-111 aircraft and a section of the T-39 wing box. These two components had the same basic concept as other early structural components, thin composite skins bonded to honeycomb substructure. The early applications of composite materials thus employed membrane states of stress only.

The design and fabrication of composite components with different and more complex design concepts followed the initial membrane applications. To date, approximately 100 aircraft component programs have been initiated (refs. 9 and 10). These programs vary in complexity from applying composites to fins and slats to their use for wings and fuselages. Several programs have also been completed in propulsion, missile, and space system applications (refs. 11 to 13).

Weight savings for these applications range from a low of 5 percent to a high of 54 percent. However, these programs were generally conducted on a simple material-substitution basis in which the basic shapes remained the same as the existing metal designs. Increased payoffs are possible when the use of composites is considered at the initial design stage. At this point, the weight savings can be reinvested in increased performance, range, payload, or operating economy.

2.1 Materials

Generally, the design engineer considers tailoring of composites only in terms of material strength or stiffness. However, many other characteristics can be tailored, including fatigue, thermal expansion, fracture and chemical resistance, electromagnetic transmissibility, and damping. Consideration of all these characteristics can greatly increase the design engineer's flexibility in material selection.

2.1.1 Material System Design

The basic constituents of a composite material system are fibers or filaments embedded in a matrix. A fiber-glass cloth (scrim cloth) is sometimes used as a carrier system and the fibers are sometimes treated with a special coating. Sometimes polymer matrices contain residual solvent and curing agents. These and other parameters constitute the variables in the design of a material system. Commercially available composite systems are described in the following text.

2.1.1.1 Basic Constituents

The filaments most commonly used in composite structures are boron, Borsic© (silicon-carbide-coated boron), various types of graphite fibers, and high-modulus polymers such as PRD-49. These fibers are characterized by the properties shown in figure 1 (ref. 14).

The Borsic© and graphite fibers are examples of how materials can be tailored to meet specific requirements. When boron fibers were first used with an aluminum matrix, the processing caused dissolution of the boron in the aluminum. The problem was solved by coating the boron fiber with silicon-carbide. Graphite fibers are made by graphitizing tows or bundles of organic precursor filaments. The modulus and strength of the fibers depend primarily upon the graphitizing temperature. Thus, three types of graphite filament are currently produced with varying moduli (high, medium, and low).

High-performance organic fibers such as PRD-49 exhibit twice the modulus of glass fibers, nearly equal tensile strength, and approximately half the density (fig. 1). Current problems include a loss in properties above 423K (300°F) and low compressive properties when used in a composite (ref. 15). Although PRD-49 is not currently in direct competition in modulus with the boron and graphite fibers, it has the potential to fill the existing gap between these two fibers and fiber glass.

Epoxies are the most commonly used organic matrix resins. They offer a wide range of processing conditions, internal strengths, and adhesive properties. The epoxies used in composite-material systems are modified versions of commercial resins in which chemicals, or fillers, or both are added to enhance the basic resin's toughness and strength at high temperature. An elementary discussion of resin matrices can be found in reference 16.

High-temperature polyimide, another type of organic resin, is rapidly gaining acceptance because of recent improvements in processing characteristics (resin flow, resin cure, etc.). These improvements are being achieved by synthesizing the basic resin to obtain the best combination of strength and processability.

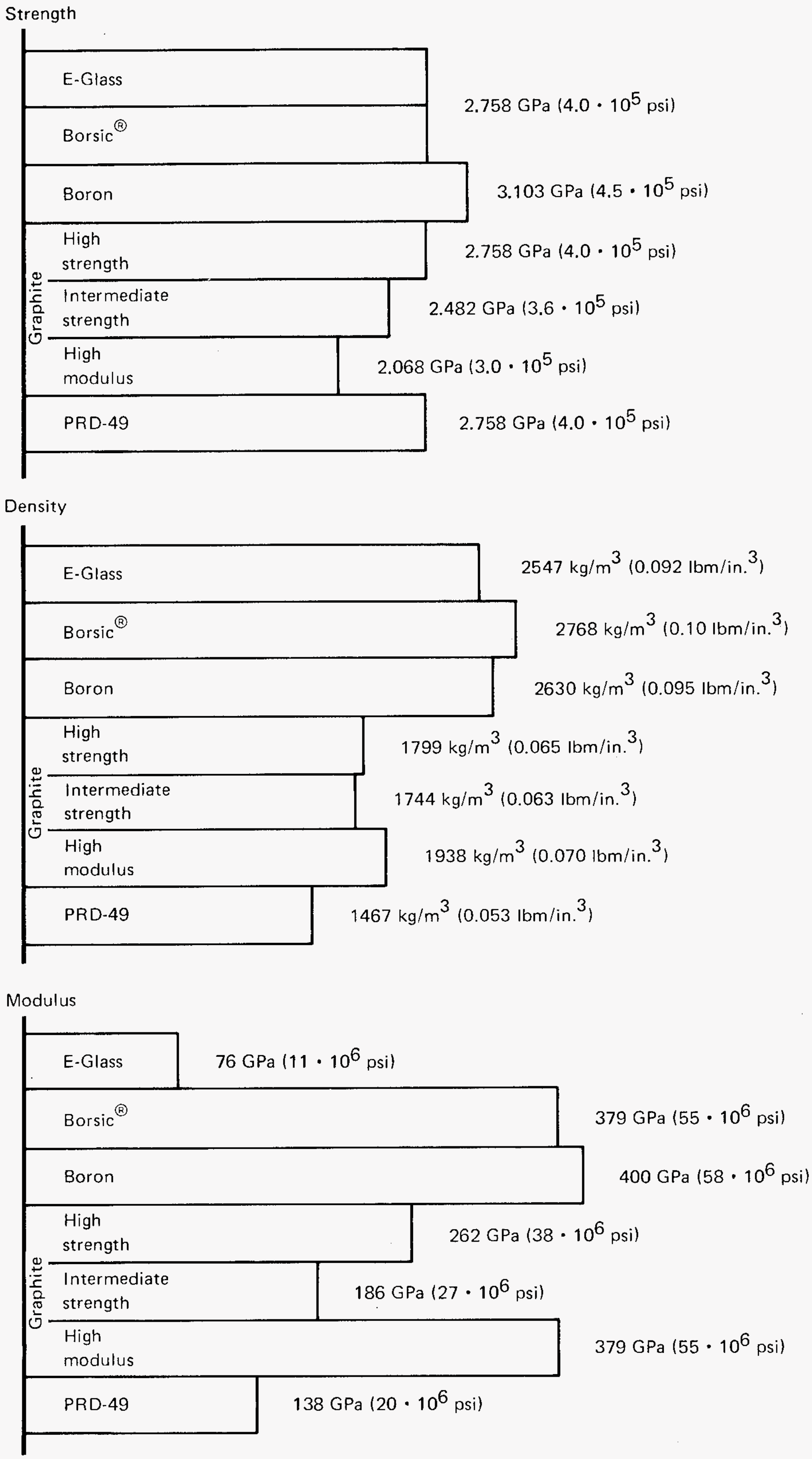


Figure 1. – Fiber properties.

Other new organic resins include phenolics and polyaromatics. However, they are still considered to be in the development stage. When preimpregnated materials (the uncured laminating material with a single ply of fiber and resin) are made with these resins, the product is often boardy or stiff and exhibits little or no tack (the ability to stick to itself or the tool upon contact). In processing, the product exhibits low flow and excessive release of volatiles, and generally results in a weak laminate.

Aluminum alloys (6061, 2024, and 713) are the only metal matrices commercially available. However, a fiber-matrix interface problem can result from high processing (diffusion bonding) temperatures [756K to 812K (900° to 1000°F)] or pressures [up to 41.37 MPa (600 psi)].

The general characteristics of the epoxy, polyimide, and metal matrix systems are listed in table I (ref. 14).

2.1.1.2 Systems

Combining fibers and matrices into specific material systems is not always a simple, straightforward process. Success depends on understanding the role of the interfacial bond between the fiber and matrix in the end product. In resin-matrix composites, it is generally desirable to strive for a high-strength interfacial bond; in metal-matrix composites, strength is often enhanced by a weak interfacial bond. However, the general approach has necessarily been empirical because little is known of the parameters governing interfacial bond strength.

A standard laminate orientation code has been established in reference 14. The following example is used for illustration: $[0/\pm45/90]_s$. The subscript "s" implies a laminate that is symmetric about its midplane. The ply arrangement for this example is as follows: 0° $+45^{\circ}$, -45°, 90°, 90°, -45°, $+45^{\circ}$, 0°. Where "c" is used in place of "s," it indicates a class of laminate. A number subscript after an angle indicates the number of plies of that orientation.

Simple experimental techniques [such as the acceptance-type flexure tests, $[0]_c$ and $[90]_c$ flexure tests (ref. 17), and horizontal shear tests (ref. 18)] and photomicrographs of failure surfaces have identified several key parameters affecting the lamina strength (ref. 19). Among these are fiber strength and dispersion, matrix and fiber moduli, fiber spacing, volume fraction (fiber content), void content, and the chemical and thermal characteristics of the fiber and matrix. Analytical techniques such as bounding, classical elasticity, and bundle theory support these findings. Other parameters such as fiber size and geometry, surface preparation, coatings, and processing variables also affect the lamina strength.

TABLE I.–GENERAL CHARACTERISTICS OF MATRIX SYSTEMS

Matrix material	Maximum service temperature range	General Characteristics
Modified epoxy	450K (350°F) continuous 489K (420°F) intermittent	Thermosetting resin utilized for low-pressure cure [approximately 6.4 MPa (100 psi)] laminate requiring a minimum of a 450K (350°F) cure for 450K (350°F) service applications
Polyimide	589K (600°F) continuous 644K (700°F) intermittent	Thermosetting resin utilized for low-pressure [approximately 13,8 MPa (200 psi)], laminating at a 450K to 589K (350°F to 600°F) cure plus an extended postcure. The polyimide resin family is characterized by difficult processing, good dielectric properties, and low cured-laminate outgassing
Aluminum	589K (600°F) continuous 644K (700°F) intermittent	6061 and 2024 alloys generally require press diffusion bonding with vacuum at about 772K (930°F), and under 206.8 to 413.7 MPa (3000 to 6000 psi) pressure to consolidate the composite 713 aluminum braze alloy requires 839K (1050°F) under vacuum and about 6.9 MPa (100 psi) pressure to consolidate. (Used with Borsic® filaments only)

Other parameters that are difficult to characterize evolve from the processing of the material system. For example, in the fabrication of an article with preimpregnated material, it is desirable that the prepreg have a given tack controlled by the degree of resin advancement. However, an exact relationship between the degree of tack and quality of laminate has not been quantified. The degree of resin advancement also affects the amount of resin that can be bled from the laminate and consequently affects the cured laminate thickness and possibly void content. Voids occurring in the laminate from entrapped air and volatile gases have been shown to reduce strength significantly. Other variables affecting laminate strength are fiber collimation, fiber-matrix interaction, and plasticization (e.g., softening of the resin due to chemical reaction with water) (see ref. 18).

The maximum strain capability of a material system is important, and determines the use of certain systems for particular applications.

2.1.2 Material Design Levels

Structural design begins with a characterization of the response of the material to static and dynamic loads in the expected environments. This material characterization can proceed at three basic levels: (1) the constituent level, with its associated micromechanical analyses, (2) the lamina, or ply, level, with its associated macromechanical analyses, and (3) the laminate, or assembly of plies, level, with its empirical approach. In each case, the end goal is to determine laminate strength and moduli. Only the last two levels are in general use. The reason for this is that the micromechanical analysis models, although more sophisticated than those at the lamina level, introduce requirements for specific constituent properties such as transverse strength and modulus of a fiber and *in situ* matrix properties which can only be estimated from lamina or laminate data. In addition, assumptions must be made regarding such parameters as fiber spacing, fiber cross-sectional geometry, nature of the fiber-matrix interface, and uniformity of microstructure. However, micromechanical analysis can often explain the contributions of individual constituents to the structural integrity of the lamina (ply); it can also help identify the predominant mechanisms that initiate failure.

Of the two commonly used design levels, the laminate level does not lend itself to an understanding of the failure mechanism and therefore restricts the engineer from optimizing his design. This procedure is also expensive because of the extensive characterization required for each laminate.

An advantage of lamina-level design is that only the basic lamina need be characterized and then, if necessary, spot checks can be made on the laminate to confirm the predicted strength. Another advantage is that it is easily adapted to optimization of the laminate ply orientation.

Once a decision has been made as to the level of material characterization (lamina or laminate), the next step in the design process is to define a laminate design criterion, which includes definitions of design limit and ultimate stresses for the laminate and an applicable failure theory.

When an engineer selects a failure theory with which to predict laminate strength, he will inevitably select one previously developed from a study of plywood, metals, or single crystals (refs. 20 to 28). Figure 2 compares several of these theories for the boron-epoxy material system. These theories are still academic because supporting biaxial test data are inadequate for their evaluation (refs. 29 to 31).

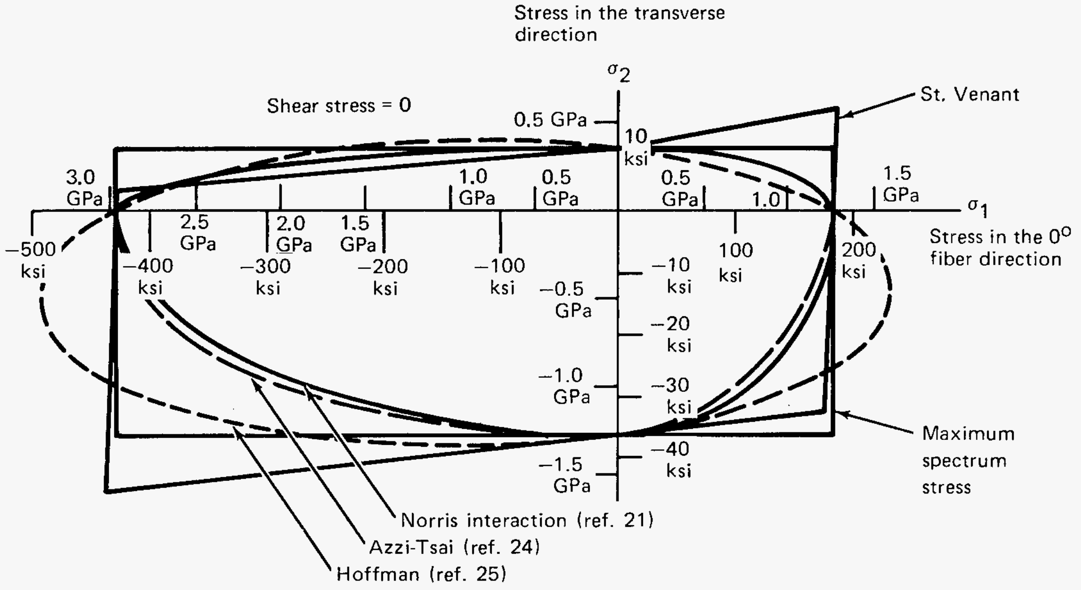


Figure 2. – Lamina failure surfaces for boron-epoxy.

Although the definitions of limit and ultimate loads in reference 14 are generally accepted, opinions differ on their application to laminate characterization, specifically, to allowable stresses. Several approaches are discussed in reference 10. Two which have received wide recognition are (1) the maximum-strain criterion (ref. 32), and (2) the modified maximum-stress criterion (ref. 33). The principal differences between these two criteria are as follows:

- The first criterion is defined at the lamina level while the second is defined at the laminate level.

- The first criterion defines the ultimate allowable stress as the maximum laminate stress attainable without the rupture of any lamina. The second criterion defines the ultimate allowable stress as the maximum laminate stress attainable without the rupture of the laminates, and neglects the contribution of lamina transverse tensile failure to laminate strength.

Figures 3 and 4 present the theoretical design limit interaction curves for the $[0/\pm45/90]_s$ and $[0_2/\pm45]_s$ laminates developed with these criteria. The greatest differences in the two criteria occur in the compression-compression quadrant. Often, however, structures loaded in compression are critical in buckling, and these differences become immaterial.

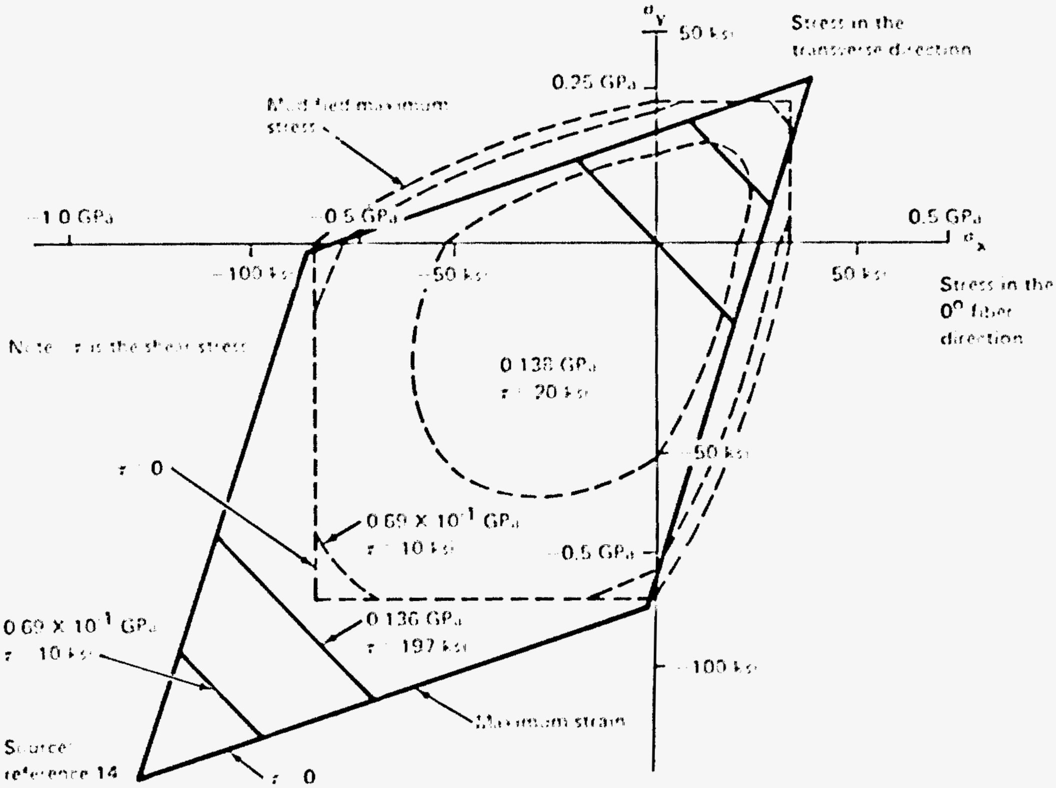
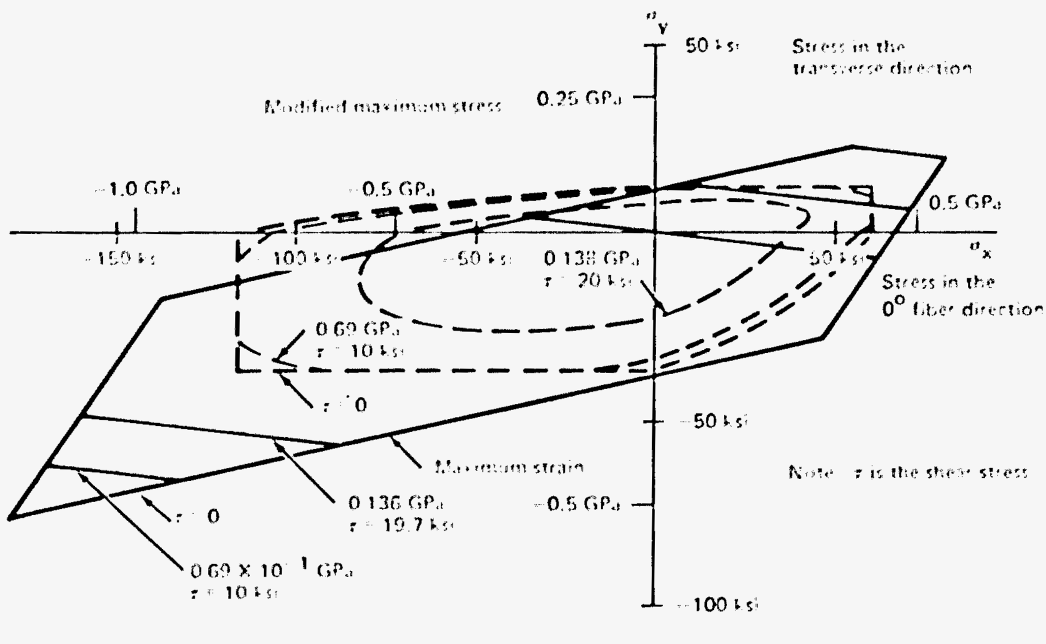


Figure 3. – Limit interaction curves for $[0/\pm 45/90]_s$ boron-epoxy.



Source: reference 14

Figure 4. – Limit interaction curves for $[0_2/\pm 45]_s$ boron-epoxy.

Experimental data (refs. 30 and 34 to 36) indicate that laminate strength depends upon the stacking sequence, the effect of which has been attributed to interlaminar normal and shear stresses at or near the boundaries (refs. 37 and 38). Since lamination theory does not account for these interlaminar stresses, current allowable-stress criteria are incomplete. However, it has been demonstrated that lamination theory adequately describes the stress field in regions removed from the boundary (refs. 39 and 40). Thus, the problem of interlaminar stresses becomes acute only at boundaries and discontinuities. Fatigue data support this conclusion (refs. 34 and 36). To date, attention has been given only to the problem of the free edge, such as a cutout (ref. 37). The role of interlaminar stresses in the behavior of a loaded edge (e.g., a bolted joint) has received little or no attention; therefore, laminate properties are generally confirmed experimentally with spot checks.

2.1.3 Material Characterization

The mechanical and physical parameters needed to describe the behavior of a composite material system under various loading or environmental conditions are defined and current statistical methods and test procedures are treated in the following text.

2.1.3.1 Statistical Design Data

The statistical method for determining static design allowables for composite materials is the same as that used for conventional structural materials (ref. 41); that is, the Mil-Hdbk-5B definitions of A and B values have been applied to static mechanical properties. Usually, the only data developed statistically are tensile data in the directions of the lamina (laminate) orthotropic axes.

It has been proposed that other properties such as compression, shear, and bearing strength be established by the same ratio of mean to allowable values as the tensile data (ref. 14). This procedure may be adequate for boron-epoxy systems (ref. 42), where the compressive strength is significantly greater than the tensile strength and where differences in the coefficients of variation of the two failure modes (tension and compression) are not important. (The coefficient of variation of a normally distributed variable is its standard deviation divided by its mean.) However, for other material systems such as graphite-epoxy, where the compressive strength is of the same magnitude as the tensile strength (or less), differences in the coefficients of variation become important. Thus, the ratio procedure is not always applicable.

Current design procedures for establishing allowable or mean property values of composites for various temperatures follow those used for conventional material systems (ref. 41). Strength at temperature is represented as a curve depicting the percentage of room-temperature (RT) strength retained as temperature increases. An elevated-temperature allowable is obtained by finding the appropriate percentage and multiplying it by the RT design allowable. No statistical significance is attached to these curves. However, in most practical applications, the product of an RT design allowable value and an appropriate percentage value from the derived curve is regarded as an allowable value at the indicated temperature.

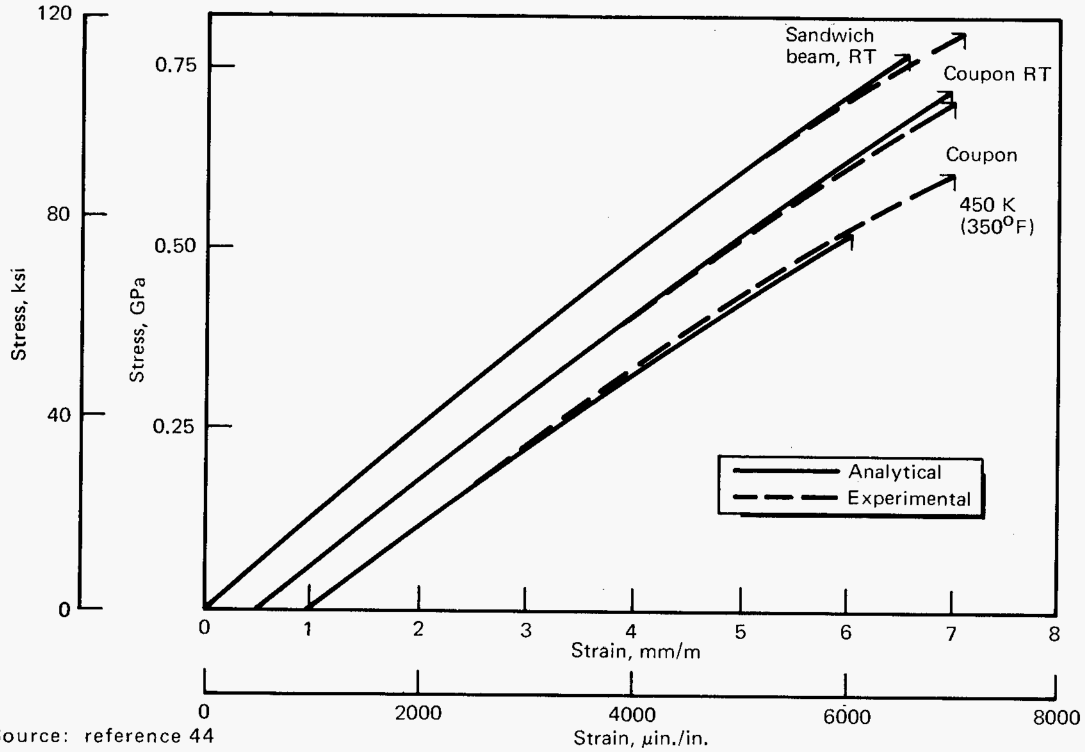
Before a percentage retention curve for strength properties is prepared, a working curve is developed with the following procedure. First, the range and average of the data points are plotted at each temperature. Next, a smooth curve is fitted through the averages at each temperature and the minimum values of the range at each temperature. The working curve is drawn in such a manner that, except for room temperature, it lies below the smooth mean curve and is no higher than 5 percent above the smooth curve drawn through the minimum values at each temperature. At room temperature, the working curve passes through the average value and is then converted to the percentage retention curve by defining the mean RT strength as 100 percent.

Since it is possible in compression strength for the failure mode to change with temperature, it may be assumed that the probability of failure also changes. The method would be based on regression analysis (refs. 43 and 44).

With lamina design allowables established, the question arises as to whether laminate strength can be predicted from a knowledge of lamina behavior. It has been clearly demonstrated that lamination theory accurately predicts laminate initial in-plane elastic properties from lamina data. With the proper failure criterion, it appears reasonable to assume that when this theory is incorporated in a nonlinear analysis (ref. 45), it can be accurately used to predict stress-vs.-strain response to failure.

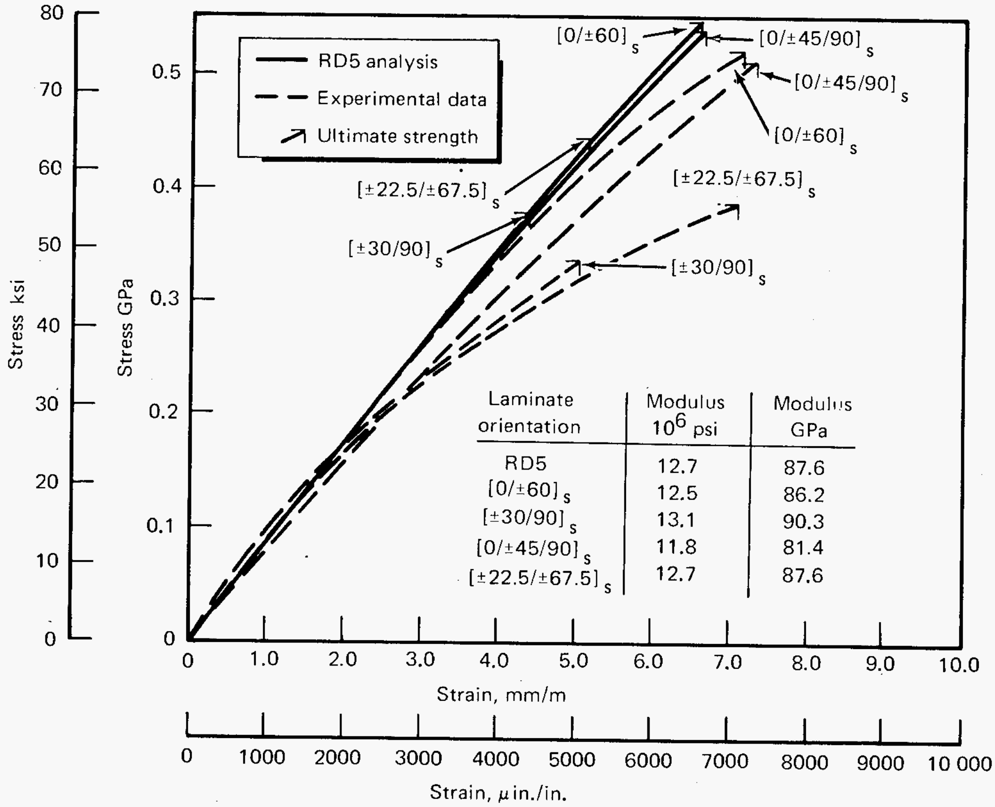
It should be noted, however, that these observations have been made from laminate tension data where the filament controlled the failure mode; that is, the resulting laminate stress-strain curves are virtually linear to failure. This relationship is shown in figure 5, which compares experimental curves for the $[0_2/\pm 45]_s$ laminate with those predicted analytically (ref. 44). [The specimen-configuration dependence (coupon vs. sandwich beam) is unresolved.]

A problem arises, however, when matrix-governed behavior (stress transverse to the fiber and in-plane shear) contributes significantly to laminate failure. For example, analysis of the two quasi-isotropic laminates $[0/\pm 60]_c$ and $[0/\pm 45/90]_c$ indicates that both should have the same elastic modulus regardless of the direction from which this property is experimentally measured. Figure 6 compares analytical predictions with



Source: reference 44

Figure 5. – Stress-strain curves, $[0_2/\pm45]_s$ boron-epoxy tension.



Laminate orientation	Modulus 10^6 psi	Modulus GPa
RD5	12.7	87.6
$[0/\pm60]_s$	12.5	86.2
$[\pm30/90]_s$	13.1	90.3
$[0/\pm45/90]_s$	11.8	81.4
$[\pm22.5/\pm67.5]_s$	12.7	87.6

Figure 6. – Stress versus strain for quasi-isotropic boron-epoxy laminates.

experimental stress-strain curves obtained from the $[0/\pm 60]_S$ laminate tested in 0- and 90-deg load conditions and the $[0/\pm 45/90]_S$ laminate tested in 0- and 22.5-deg load conditions (ref. 44). As expected, the initial moduli of the experimental data agree with the analytical value. However, figure 6 indicates that the experimental stress-strain curves do not correlate well with either the analytical predictions or with each other.

This anomaly also exists for laminates of the $[\pm\theta]_c$ family. References 46 and 47 demonstrate that the lack of correlation is not limited to the analytical method of reference 45, but exists when other analytical methods (refs. 48 and 49) are employed.

Characterization of the laminate itself in its principal axes does not ensure a knowledge of the laminate strength under every load condition. For example, if the intercepts of the failure-surface axes for the $[0/\pm 45/90]_S$ laminates are known and maximum-strain failure theory applied to the data, this criterion does not explain the failure of the $[\pm 22.5/\pm -67.5]_S$ laminate (fig. 6), which has approximately the same failing strain as the $[0/\pm 45/90]_S$ laminate. Similarly, application of the maximum-stress failure theory also fails to account for the failure of the $[\pm 22.5/\pm 67.5]_S$ laminate. The presence of lamination residual stresses affects the matrix-controlled laminate strength. For a more realistic correlation of theory and experiment, the lamination residual stresses must be considered (ref. 50).

A detailed study of these observations shows that an apparent interaction exists between the two matrix behavior modes, tension and shear (ref. 44). Since existing failure criteria separate the shear contribution from the normal components, current practice is to be cautious in utilizing strengths beyond the limit values defined by the failure criterion.

The establishment of material design allowables is not strictly limited to static properties. If fatigue is considered in design, then fatigue allowables are established. References 51 and 52 describe methods of statistical evaluation of fatigue data.

2.1.3.2 Mechanical and Physical Properties

As stated earlier, design allowables based on statistical analysis of data are usually generated for static strength properties along principal axes. The other mechanical properties (e.g., elastic moduli, Poisson's ratio, and creep strength) and the physical properties are normally presented as average or typical values. Data for composite materials have generally been presented and interpreted in the same way as those of conventional materials. However, one of the most controversial subjects in the testing of composite materials is the manner in which these data are obtained; the problem is one of specimen design.

The pros and cons of specific specimen designs are discussed in references 14, 53, and 54; in some instances, certain specimens are recommended. To date, only one test specimen design has been generally recognized throughout the aerospace industry (ref. 14); the straight-sided tensile coupon developed at the Illinois Institute of Technology Research Institute, commonly referred to as the IITRI coupon. Societies such as the American Society for Testing and Materials (ASTM) are currently striving to develop acceptable specimens for other mechanical properties.

In general, the determination of most physical properties for composite materials does not pose a problem. Methods developed for conventional materials such as those found in the ASTM Standards can be readily modified to account for the peculiarities of a particular material system. For example, the technique used to measure the resin content of a fiber-glass laminate requires burning off the resin in a high-temperature furnace. This technique does not work for graphite- or boron-epoxy because both graphite and boron also burn off with the resin and change the weight balance. Therefore, chemical etching is used.

The diversity of available composite material systems is enhanced by the freedom the designer has in selecting a laminate orientation. Thus, the task of tabulating all composite material properties becomes insurmountable. The tendency in composite technology has been to identify key unidirectional and cross-ply laminates and characterize their properties. This identification then yields design guidelines. Static design strengths, elastic properties, physical constants, fatigue, and environmental effects for the more common material systems are presented in reference 14. Additional fatigue data can be found in reference 55.

2.2 Design

The primary objective of the design process is to relate experimentally derived properties of materials or structural elements and engineering principles to functional requirements in order to obtain reliable and producible structure. The design process accounts for design conditions, design factors, scaling, reliability, analysis, producibility, and maintainability.

2.2.1 Management

In a deterministic design approach, the management plan establishes the design factors (see Sec. 2.2.3). For a reliability-based approach, a detailed management plan is prepared to establish the control aspects of the procedure described in Appendixes A and B.

2.2.2 Design Conditions

The current or historical definition of limit load is the maximum load expected to be experienced by the structure in its operating environment during its service life. For a deterministic design approach, the ultimate design load is then obtained by multiplying the limit load by a standard ultimate design factor (usually 1.5 for aircraft, 1.4 for manned space vehicles, and 1.25 for unmanned space vehicles)-(see refs. 56 and 57). This factor is intended to allow a sufficient margin to ensure safe operation during a rare excursion above the maximum expected load. Sufficient data are now being recorded to allow a more rigorous treatment of this overload requirement. For a reliability-based design approach, the overload condition is set from flight statistics (e.g., a once-per-fleet-life load level).

A random fatigue design spectrum including environmental conditions is sometimes defined and used throughout the design process to eliminate dependence on a cumulative damage theory. Thus, fatigue requirements may be satisfied during the design phase rather than being checked after the design is rigidly defined (e.g., ref. 58).

2.2.3 Design Factors

Uniform factors of safety, coupled with statistically determined material allowables, have come into use during years of structural experience with metals as a means of ensuring structural integrity at some undefined confidence level. These factors have been applied to composite structures; however, they should not be applied automatically to composite structures for the following reasons:

- Limited experience with composite materials (refs. 59 to 61)
- A demonstrated scale effect from coupon to component (refs. 62 and 63)
- The brittle nature of certain failure modes
- The inapplicability of current cumulative damage theories

Ultimate design factors commonly range from 1.2 to 2.0 for general structure. They are used to cover uncertainties and variations in material properties, loads, and manufacturing processes. In addition, current practice calls for fatigue or scatter factors ranging from 2 to 4.

2.2.4 Environment

Environmental conditions have an important effect on the static and fatigue strength of composite structures. Several potential problem areas are being investigated.

2.2.4.1 Corrosion

No evidence of corrosion has been reported for either boron- or graphite-epoxy systems. However, galvanic-cell reactions have occurred in which aluminum was attacked when attached to graphite-epoxy laminates (ref. 64). No corrosion problems have arisen in laboratory tests with hydraulic fluid, solvents, fuel, and other chemical agents normally found in the vicinity of aerospace structures, although epoxy paint remover is detrimental to epoxy-based composites. Some chlorinated oils will attack epoxy matrices.

2.2.4.2 Long-Term Stability

Long-term environmental problems have not appeared because only a few parts have seen more than three years of service or approximately 1000 hours of flight time. Certainly, the experience gained with bonded structure indicates potential problems. Experiments show that the absorption of water vapor degrades the strength of epoxy composite systems (ref. 65). The degree of degradation is a function of the resin formulation and cure. Strength loss of up to 60 percent has been reported. Various coatings have been tried, but none has been successful. However, these data show that the original strength is regained with the elimination of the water vapor. Present practice is to minimize this problem by careful resin selection and by curing at 450K (350°F) or higher temperatures.

Epoxy systems will decompose with time when exposed to ultraviolet radiation. However, painting or pigmenting the resin prevents this decomposition.

Certain elements of some epoxy polymers sublime with time in a hard vacuum in a process called outgassing. Careful design of the resin system may hold outgassing to an acceptable level.

2.2.4.3 Abrasion/Impact

Composite systems selected on the basis of high strength, such as boron or high-strength graphite in an epoxy matrix, possess some resistance to impact and excellent resistance to abrasion. When impacted, the laminate either deflects and springs back or fractures (ref. 66).

Systems using high-modulus graphite fibers tend to show increased brittleness and thus increased susceptibility to impact damage. However, they are sometimes used in protected low-stress applications where the extra stiffness is required. Rain erosion is a serious problem for epoxy-based composites, and no adequate coating is known. Metal foil is generally used for protection in critical areas. A thin-glass cloth has been used on the outer surface of the laminate in less critical areas.

2.2.4.4 Lightning

Both boron and graphite have a high electrical resistance. Epoxy is a good insulator. Thus, if any significant quantity of electrical energy enters a laminate, the fibers involved get hot and break, disbond, or burn. A lightning strike will burn a hole in a plastic-based laminate and may overheat the fiber some distance from the visible damage. Lightning protection systems must prevent entry of the electrical energy into the laminate. Approaches now being developed use an aluminum or titanium foil or sheet around the periphery that is capable of shunting a full-intensity strike, as well as a conductive surface coating capable of shunting lower-intensity strikes (ref. 67). Coatings of conductive paint or aluminum screen wire have been used, as have aluminum foil strips.

2.2.4.5 Heat

Both boron and graphite fibers retain their strength well above the limiting temperature of current plastic matrix systems. Metal-matrix systems may be used slightly above the normal operating temperature of the matrix. Matrix-governed failure modes are important at or near the limiting temperature for the material system.

Available epoxy systems possess long-term stability at 450K (350°F). Polyimide systems are being developed for use up to 589K (600°F). Creep and stress rupture tests in fiber-controlled failure modes show excellent retention of initial properties (ref. 10). Matrix-controlled failure modes show significant time-dependent strength and deformation effects. However, overheating by such means as exposure to fire does not degrade the laminate until actual charring or delamination occurs. The progressive delamination process retards damage because of its insulating effect. Resin systems in general use will pass standard ASTM flammability and toxicity tests.

2.2.5 Scale Effect

The current or metals-oriented criteria are based on an assumption that coupon and element strength translates directly into the strength of a complex structure. However, bonded joints in composites have demonstrated a significant scale effect (refs. 62 and 63). Such joints demonstrate lower fatigue life and lower static strength with increasing joint size. Further, the scale effect differs for fatigue and static strength and differs with configuration. Scaling is done at four levels: (1) material, (2) structural elements, (3) subcomponents, and (4) components. Failure data for each level are determined as the design progresses.

2.2.6 Reliability

Although general reliability practices are defined in the literature (refs. 68 to 79), these concepts are rather new and have seldom been used on structures. In particular, reliability-based design procedures have never been implemented on a production basis for composite structures.

Attempts to provide high reliability through use of arbitrary design factors have often been unsuccessful (refs. 62, 80, and 81). The reliability assessments have usually been made only after sufficient service data have become available for the structure. For instance, reference 80 shows that of the total fleet of 2292 F-100 airplanes, 379 had major accidents that were attributable to material failure. Thus, the probability of fleet survival was 0.835, a value that would certainly not be acceptable in design.

The use of composite materials has created a need to move from traditional design procedures to more rigorous ones. Design procedures are needed to account for structural strength scatter and fleet size. These procedures, which must be based on subelement tests subjected to the fatigue design spectrum, would ensure structural integrity throughout the service life. Several companies are now formulating and evaluating reliability-based design procedures for composite materials.

2.2.7 Analysis

The design of composite structures is generally integrated with the analytical effort, requiring the coordination of materials, design, and fabrication knowledge. However, the stiffness, load paths, and stresses of composite structure are so different that many changes must be made in conventional design practice. For example, the anisotropy of the material requires the structural designer to interact with the related disciplines of aerodynamics and control to a greater degree than with conventional materials. Similarly, cooperation between the designer and the fabricator is required because of the limited inventory of proven manufacturing techniques.

Composite materials offer a design team the opportunity to design and fabricate a structure with the desired directional properties. The complexity of the analysis and the wide range of element parameters available to the design team have brought increasing use of iterative computer techniques. Composite designs require that attention be paid to structural optimization, configuration tradeoff studies, producibility, and reliability.

Design of a composite structure normally proceeds through several iterative phases. The initial phase is a concept evaluation study to determine the overall design configuration. Parametric studies on various vehicle configurations and geometric constraints produce the initial optimized designs. Preliminary design studies further refine and develop structural concepts through analysis and testing of small-scale components. At this point, allowances are sometimes included to facilitate repairs during operations. Such allowances, which are required by some agencies, assume the form of increased thickness or edge distance for fastener holes.

This preliminary design effort includes material performance. Weight reduction is the most important factor in selecting composite materials because of its direct effect on the design indices of cost-to-weight and strength-to-weight. The relative merits of various combinations of materials and configurations are usually analyzed with computers to obtain a minimum-weight design.

The analytical phase of the design process includes stability and stress analyses for the various loading conditions and environments. Sandwich construction, stringer-stiffener construction, bonded and mechanical joints, and various miscellaneous details such as corners, cutouts, attachments, and intersections are taken into account at this point in the design. Both laminate thickness and filament orientation can be varied to satisfy stress or stiffness requirements. Reference 82 is an authoritative review of detailed analytical and design information on composite applications. Appendix C lists rules of thumb for the detail designer.

2.2.7.1 Internal Load Determination

An accurate in-depth structural analysis is needed to determine the characteristic sensitivity of a composite structure. The necessary depth and accuracy are usually obtained with digital computer programs. Automated analytical procedures are applicable to the design of most major structural components. Most of these procedures center on the finite-element simulation of the composite component; an example is NASTRAN, a general-purpose digital computer program designed to analyze the behavior of elastic structures under a wide range of loading conditions using the displacement method (ref. 83).

Numerous computer programs that are similar in approach and capability to NASTRAN are available throughout the aerospace industry. These programs have anisotropic and orthotropic finite elements to analyze any structural component fabricated with laminated composite materials. Both linear-elastic and nonlinear constant-stress finite-element programs are available. Reference 84 describes a nonlinear anisotropic finite-element program which uses the stiffness matrix method of analysis and the fourth-order Runge-Kutta forward integration scheme to set up and solve the system of nonlinear ordinary differential equations.

2.2.7.2 Laminates

More complicated analytical methods are used with fiber-reinforced composites than with conventional isotropic metallic structures because they are inherently orthotropic or anisotropic materials. The influence of material orthotropy on stability, stress, and dynamic response is taken into account in the design effort.

Typically, a fiber-reinforced composite laminate is balanced about its principal axis and mirror-symmetric about its midplane: it is therefore essentially orthotropic. The various types of laminates are described below:

Isotropic – Elastic properties the same in all directions; metals are usually considered isotropic.

Orthotropic – Orthogonal properties, different in principal directions.

Anisotropic – Unequal properties, tension load will cause in-plane shear.

Coupled – Orthotropic or anisotropic and not symmetric about the midplane.

The differences between these types are illustrated in figure 7.

An orthotropic laminate behaves anisotropically when it is loaded about an axis other than a principal or orthogonal axis. The bending-stretching coupling present in an unbalanced laminate causes warping due to in-plane strains, which may be induced by in-plane loads or thermal stress due to different coefficients of expansion between the different plies. These strains and stresses may be computed by procedures such as those given in reference 85.

Laminate properties may be used in combination with a failure criterion such as St. Venant's maximum strain theory (Sec. 2.1.2) to obtain a design limit surface, as shown in figure 8 for a typical composite laminate. Current analytical techniques utilize lamination theory, which allows the complete laminate constitutive relationships to be derived from basic lamina properties (ref. 86). Limit design criteria are used to calculate margins of safety when limit loads are compared with design limit strains or stresses. The design limit surface is used by the designer as an allowable interaction curve. A separate curve is generated for each laminate orientation and temperature. Automated procedures such as procedure SQ5 (ref. 85) are utilized to generate these interaction curves. An interaction diagram for average in-plane stresses is three-dimensional, but it can be depicted in two dimensions with the third variable τ_{xy} appearing as cutoff lines. This failure theory is applicable only to the linear portion of material havior.

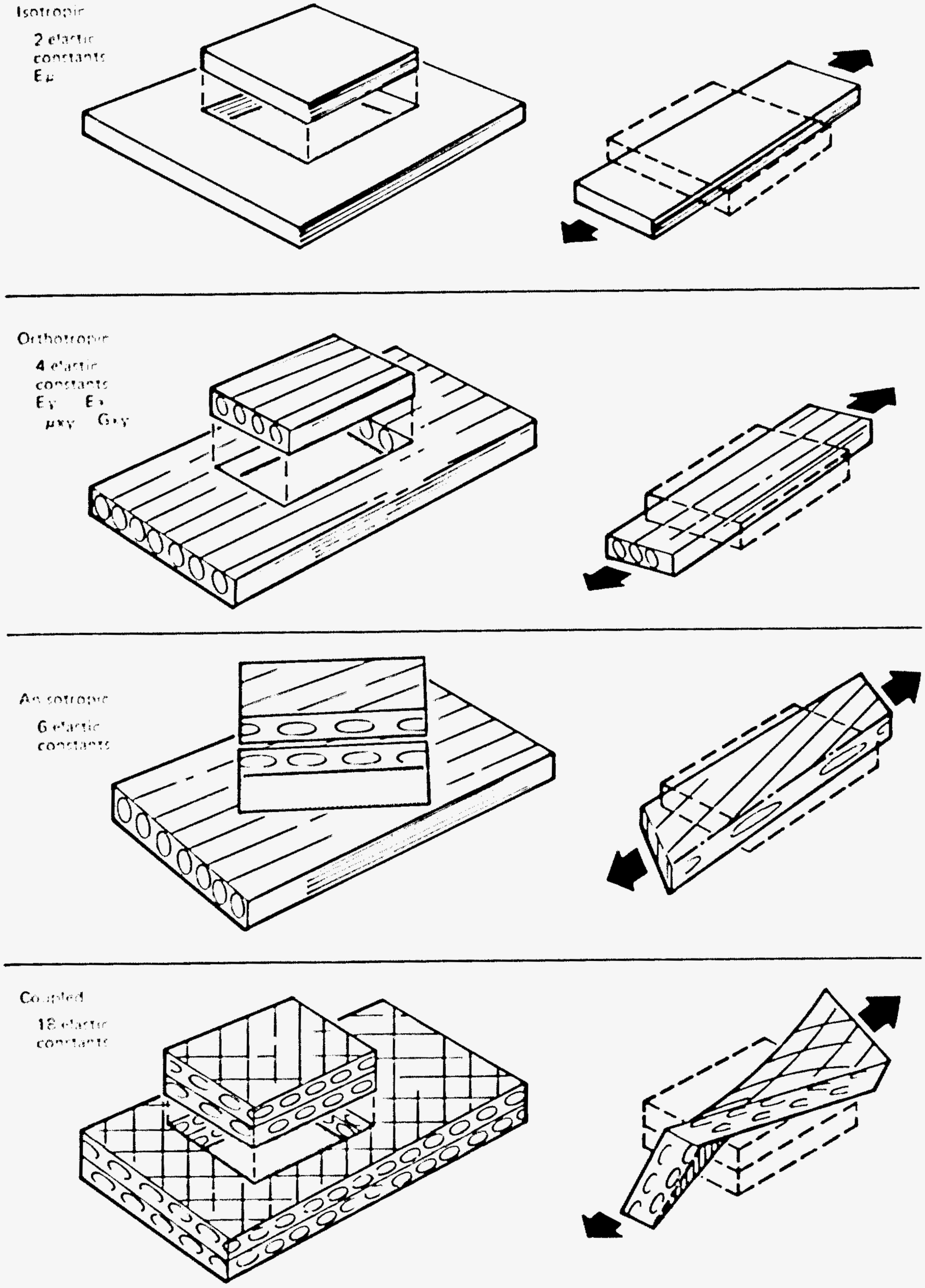


Figure 7. – Composite terminology.

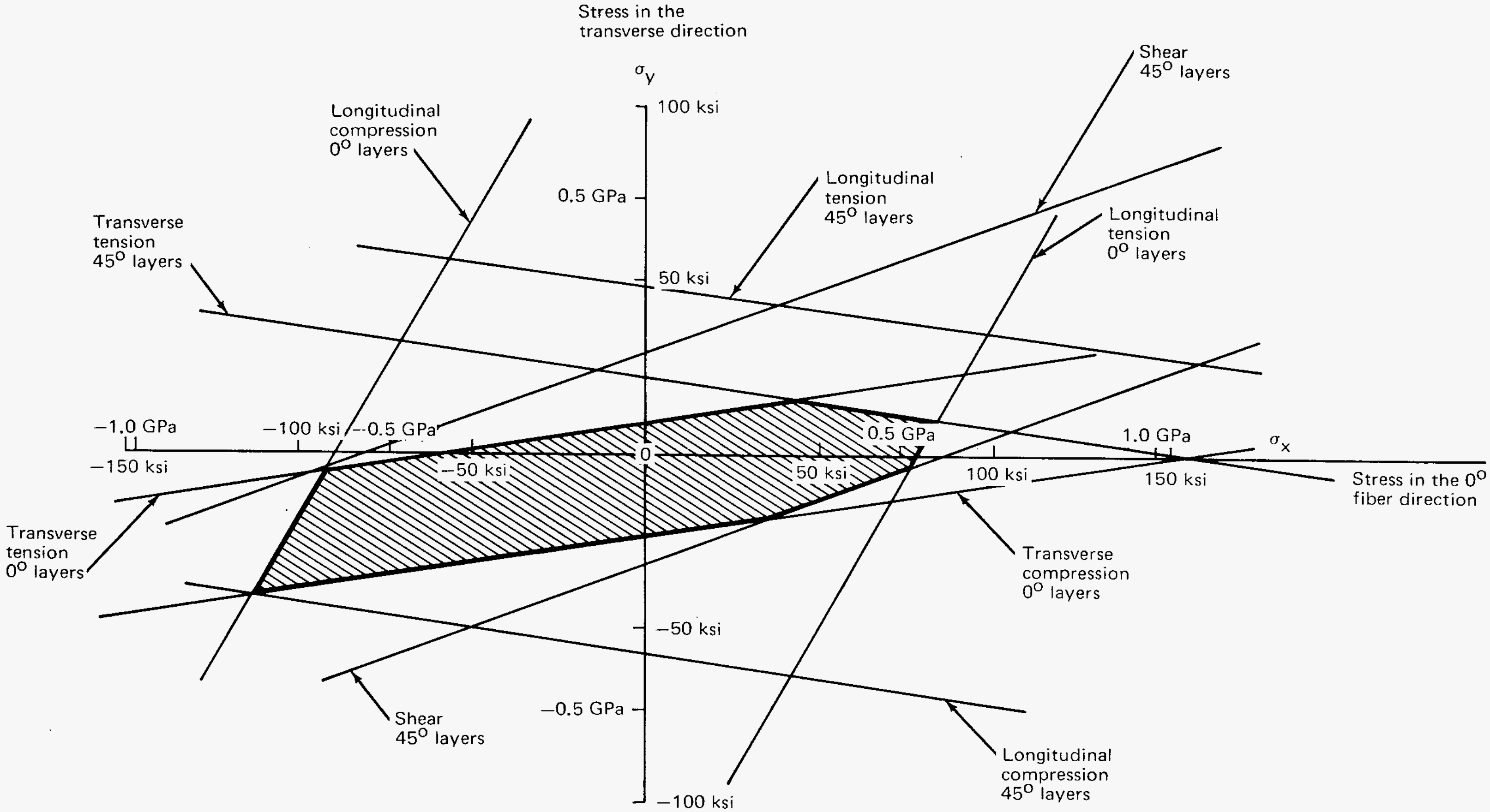


Figure 8. – Design limit surface, $[0_3/\pm45]_s$ graphite-epoxy.

If the laminate property prediction methods are extended to failure through nonlinear analytical techniques, material failure strength can be compared with the design limit strength, as shown in figure 9. Program RD5 is an example of a mechanization of this method (refs. 45 and 87).

Values of Poisson's ratio are large for some laminate families, and can create high interlaminar stresses in an otherwise uniformly stressed laminate when bonded to other materials. For example, a metal splice plate failed prematurely after it was bonded to a $[\pm 25]_c$ composite laminate (ref. 82). The splice plate provided a lateral restraint on the adjacent composite material because of the large difference in Poisson's ratio.

Composites can be tailored to meet specific strength or stiffness requirements. For example, when the design requirement is axial strength or stiffness, most of the material is unidirectionally oriented. (If the material is enclosed in some restraining member, all of the material can be unidirectionally oriented.) A nominal amount of transverse reinforcement is normally provided to account for possible off-axis loading. To satisfy shear-loading or stiffness requirements, most of the material is oriented at ±45 deg to the longitudinal axis.

In most applications, a laminate must withstand combined loading conditions. Because composites can be tailored, an optimum family of laminate orientations can be designed to satisfy a number of simultaneous requirements for membrane and bending loads and stiffness. The strength and stiffness requirements often vary widely from point to point in a structure. When this occurs, automated optimization routines are

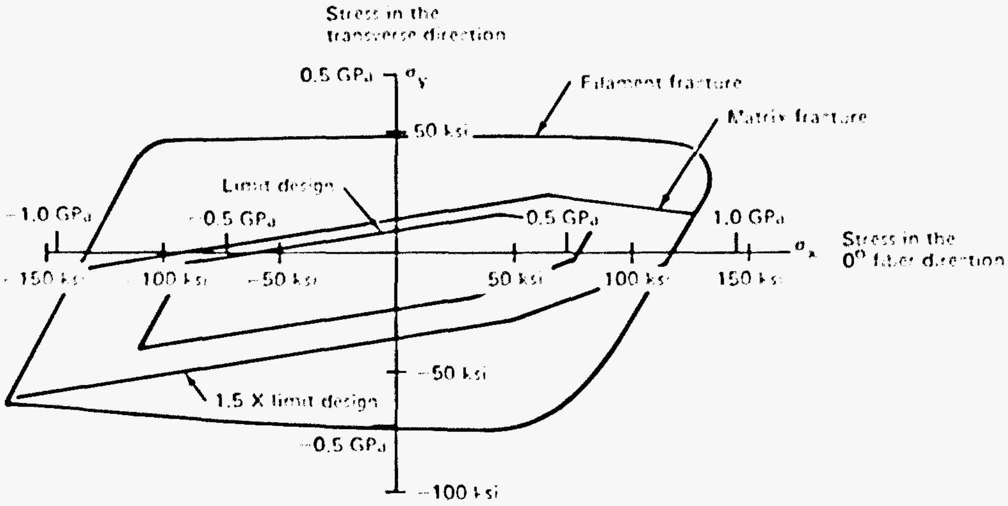


Figure 9. – Limit and ultimate interaction curves, $[0_3/\pm 45]_s$ graphite-epoxy.

useful in designing a laminate with an orientation and a stacking sequence that satisfy both structural requirements and fabrication constraints. An example of an automated procedure that provides a minimum-weight structural optimization capability for a class of anisotropic plate structures is program ATO (refs. 88 and 89). Many other programs are also available (ref. 82). This is a rapidly changing field and no attempt is made here to list all the available programs.

Generally, laminates contain at least three different orientation angles, $[0 \pm 45]_C$, to minimize matrix-governed failure modes. A single exception is the $[\pm 45]_C$ laminate used in pure shear applications. A laminate with as few as three plies (two plies for shear) has been successfully used on sandwich construction. Otherwise, minimum gage is usually six plies (four for shear) to balance the laminate and minimize warping.

2.2.7.3 Panels

Flat or slightly curved panels are used frequently in space-vehicle structures, in the form of sandwich, unstiffened, and stringer-stiffened plates. Stability considerations usually determine their major characteristics, such as the composite thickness and the filament orientation. The behavior of the panel under in-plane compression and shear loads is of primary interest (i.e., the determination of the load at which buckling occurs and the load at which collapse or cripple is induced) (ref. 2).

Basic equations of the theory governing the extensional and flexural behavior of laminated panels under small deflections are discussed in references 2, 82, and 86. The theory is based on the Kirchhoff-Love hypothesis regarding deformation. Thorough treatments of laminated plates can be found in references 90 to 93. Basic theoretical procedures for the analysis of plate stiffness depend on the coupling between membrane and flexural behavior. Stacking sequence, degree of bending anisotropy, and bending-stretching coupling are important variables. The governing equations for plate buckling are usually those for an orthotropic laminate that is midplane symmetric or a laminate in which the coupling effects are small.

References 93 and 94 present data on homogeneous anisotropic and orthotropic plates with negligible coupling. Experimental and comparative theoretical studies of the buckling of fiber-reinforced plates (refs. 95 and 96) offer insight into membrane-flexure coupling when the coupling is not negligible. Reference 92 indicates that the Rayleigh-Ritz method together with lamination theory allows accurate prediction (within 10 percent) of the buckling loads of composite plates for a variety of boundary conditions.

For plates with coupling between extensional and flexural deformations, references 90, 91, and 97 to 100 offer guidelines for obtaining reasonably accurate solutions.

Reference 101 presents both the theoretical development and pertinent example solutions for the effects of bending anisotropy, stacking sequence, and bending-stretching coupling on the buckling, dynamic, and lateral-load response of anisotropic plates. An analytical method that consists of an energy solution and makes use of the Rayleigh-Ritz technique is presented in reference 102 for an anisotropic plate. The computer program described in this reference treats the static deflection, stability, and dynamic response of anisotropic plates.

Finite-element models have been generated for composite plate systems. Membrane-type elements for elastic anisotropic plates are presented in reference 103. However, a finite-element model for extensional and flexural coupling has not been developed.

The local failure modes in composite sandwich structure are similar to those of metal structures and are fully described in reference 2, with special emphasis on the importance of the effect of low shear modulus of the core in general instability analysis.

Capabilities for analyzing postbuckling and inelastic behavior are now being developed. Preliminary results indicate that composite panels possess significant postbuckling strength (refs. 104 and 105). The impact of local stiffness on postbuckled behavior can be significant. Tests are usually conducted to confirm the design.

2.2.7.4 Shells

The design and analysis of shell structures are more complicated than for plates. Notwithstanding the numerous attempts to predict the behavior of metallic shell structures, design-confirmation testing is still necessary.

The design of shells is dominated by considerations of their stability under shear, bending, axial compression, and external pressure. Early shell-stability analyses were based on classical "perfect shell" mathematical models. Such analyses always provide unconservative predictions of the buckling strength of fabricated shell structures. This means that the analytical results must be reduced by empirical knock-down factors relating to shell geometry.

Since practical construction cannot provide imperfection-free shells, statistical information concerning the imperfections of the shell is used to provide reliable stability predictions. Because their limited strain capability does not permit large deflections, composite shells will sustain little or no load beyond the load that initiates buckling (ref. 106).

The state of the art of shell design up to 1968 is well documented in references 1 and 107. The development of general-purpose computer programs (refs. 83, 108, and 109) made possible the classical analysis of "perfect" shells of complicated geometry. Experimental results for buckling of composite cylinders are presented in references 110 to 113.

Because of basic changes in shell-stability methodology since 1968, very few procedures have been proposed for the analysis or design of composite shells. The only case to receive any attention was the axially compressed composite cylinder with imperfections, which is treated in references 114 to 117.

2.2.7.5 Joints

Both bonded and bolted joints are commonly used to join composite materials. Brazed and spot-welded joints can also be used with metal-matrix composites. Most joint designs are sized initially from analytical data or from design curves developed from existing test data and then tested and modified as necessary. In general, analytical methods have been used to predict the failure strength of joints with only moderate success. Both the analysis and the testing of composite joints are complicated by the large number of variables in the configuration of the joint, in the type of adhesive, and in the laminate itself.

In lightly loaded structure, adhesive-bonded joints are the most efficient means of static load transfer. However, the efficiency of bonded joints is reduced at increased load levels. The two basic types of bonded joints are the lap and scarf joints shown in figure 10(a). Variations of these two basic configurations allow tailoring of a given joint design to match required geometric constraints and load levels.

Some acceptable analytical procedures for simple lap and scarf joints are presented in references 10, 117, and 118. However, the simplifying assumptions made in setting up the analytical models prevent the joint strength from being accurately determined. Empirical factors obtained from the correlation of analytical and test results can be used to develop design curves such as those presented in reference 10.

Typical composite laminates have a very low coefficient of thermal expansion in the filament direction [$\alpha \cong 0$ to 5.4 μm/m/K ($\alpha \cong 0$ to 3 μin./in./$^{\circ}$F)]. Mismatch of thermal coefficients must be considered when composites are bonded to metals because the cure temperature for the adhesive is approximately 450K (350°F). Successful composite bonds can generally be obtained with either steel or titanium [$\alpha \cong 10.8$ μm/m/K ($\alpha \cong 6$ μin./in./$^{\circ}$F)], but bonding to aluminum [$\alpha \cong 23.4$ μm/m/K ($\alpha \cong 13$ μin./in./$^{\circ}$F)] is done only with caution.

The most basic form of bolted joint is the single-overlap configuration shown in figure 10(b). Variations of the basic form include the use of multiple fasteners and varying adherend moduli and thicknesses along the splice. The main difficulty in

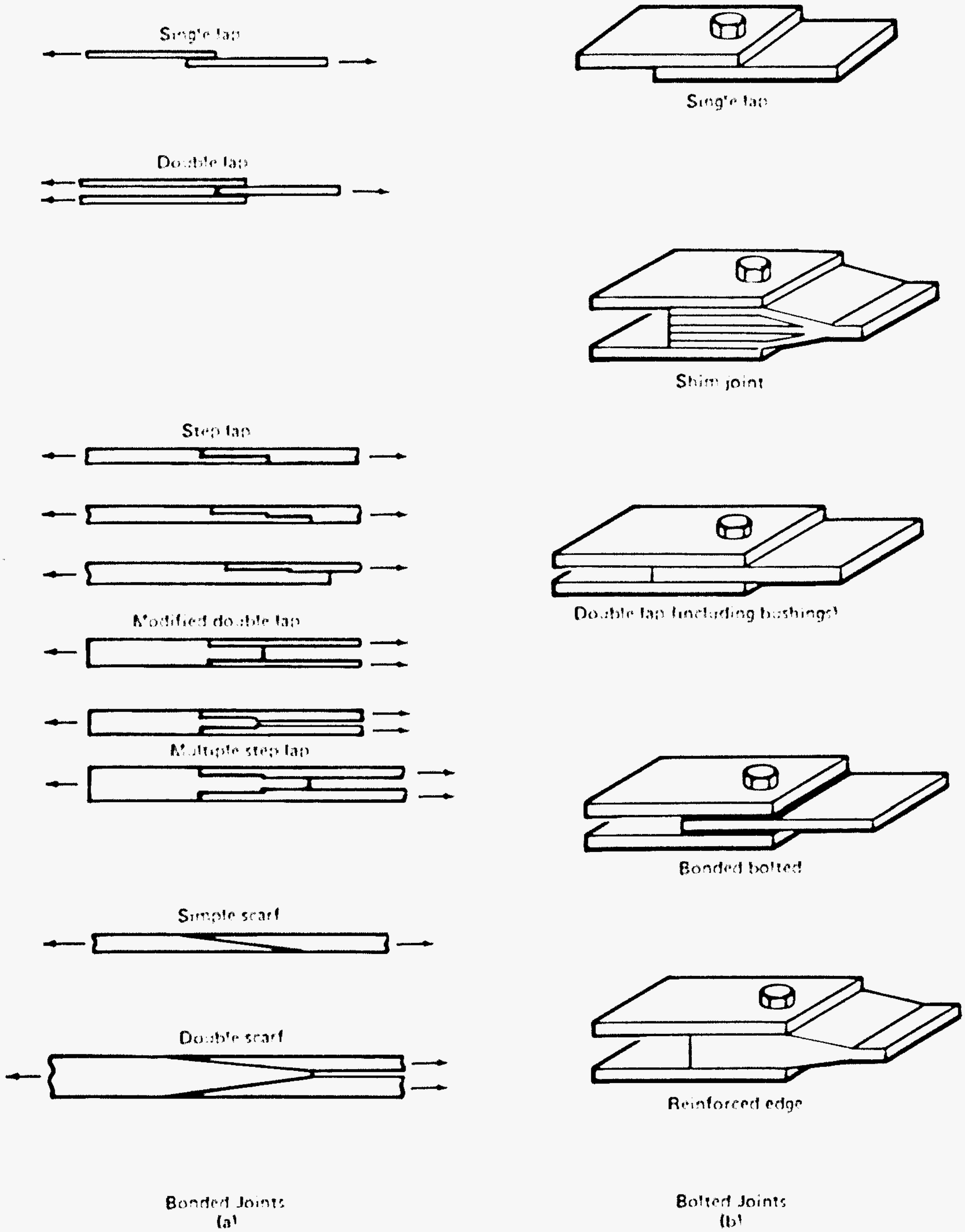


Figure 10. – Typical joint elements.

bolted-joint design is the stress concentration factor caused by the loaded hole. This can cause a net section-failure stress of 20 to 50 percent of the material allowable. The failure mode is a tension rupture originating at the point of maximum stress concentration relative to the laminate strength, which in some cases is a split along the direction of the load. One technique for reducing the loss in strength is to install the fasteners in strain-tolerant strips that are built into the basic laminate (refs. 119 and 120). The strips provide local ductility at the fastener locations, like a ductile metal.

Several finite-element analyses of isotropic bolted-joint elements have been developed with various degrees of complexity (ref. 10), ranging from a cosine distribution on the boundary of a circular hole to three-dimensional models that include bolt bending and shear deflections. One analytical technique that combines an anisotropic finite-element solution with the distortional energy failure criterion has been fairly successful in predicting the ultimate strengths in tests of boron- and graphite-epoxy joints (ref. 121). In general, correlation of analysis and test results is difficult because of the simplifying assumptions and the many possible failure modes. Analytical trends, however, combined with test data provide design curves of the type presented in references 10 and 14.

Bearing and shear-out failure modes are analyzed in the same way in composites as in metals; however, they are of second-order importance compared to the net-section failure mode.

Both static-strength and constant-amplitude fatigue data for various bolted-joint configurations are presented in references 10, 52, 122, and 123. The variables investigated include the number and size of fasteners, temperature, material type, laminate orientation, splice geometry, and stress ratio. The fatigue problem of bonded joints in composite materials is very similar in magnitude to that of metals with stress concentrations.

Fatigue-life data for a double-overlap bonded joint subjected to a random spectrum loading are presented in reference 62. One promising technique to improve fatigue behavior is to use both bolts and adhesive to transfer loads (ref. 14); however, no theoretical technique is now available for the analysis of this type of joint. Design optimization programs are being developed for multiple-fastener bolted joints and multiple-step-lap bonded joints.

2.2.7.6 Component Design for Strength and Stiffness

Extremely efficient composite components can be designed when there is one dominant design load condition. A structural element carrying only axial tension and compressions loads, for example, would be constructed primarily with unidirectional plies that provide strength and stiffness in the axial direction. The resulting composite component would have a specific strength (tensile failure strength-to-density ratio) three to four times the ratio of a corresponding aluminum member (refs. 11 and 124 to 126).

In the same manner, a composite shear web is designed with a $[\pm 45]_c$ laminate that is especially suited to provide shear strength and rigidity. Specific shear strength and shear stiffness for this application are approximately two to three times those of a corresponding aluminum shear web.

Components that carry a variety of loads such as biaxial tension, compression, and shear are designed to have acceptable strength and stiffness in all directions, much like metals. Either $[0/\pm 45/90]_c$ or $[0/\pm 45]_c$ laminates are suitable for these applications; both have specific strengths and stiffnesses about one to two times those of equivalent aluminum structures.

In designs to achieve component stiffness, specified values of bending stiffness, EI, and torsional stiffness, GJ, are maintained. In metals, once the material type is selected, these values are maintained by relocating or adding material. More options are available in designing for stiffness in composites since Young's modulus (E) and the shear modulus (G) can both be varied by changing laminate orientation. Most metals have an E/G ratio of about 2.6; but the $[0 \pm 45]_c$ family of graphite-epoxy laminates may have E/G ratios ranging from 0.5 to 7.0, as shown in figure 11. This additional flexibility permits very efficient component designs, but also complicates the design task. Hand-solution procedures can be defined for several fundamental strength and stiffness designs, but for general problems nonlinear programming techniques are more efficient (ref. 127).

Metal structures can be locally reinforced with composites to increase strength and stiffness. This is probably the most efficient way to use composites in terms of pounds of metal removed vs. pounds of composite used. References 128 to 130 discuss this "hybrid" approach.

2.2.8 Producibility

The ability to manufacture flightworthy composite structures has been demonstrated with flight-test and other research and development programs (refs. 59 to 61, 120, 131, and 132), and reference 10 contains a comprehensive section on the producibility of composite structures. However, most of these programs were one-of-a-kind fabrication exercises. The cost of these structures was many times the cost of comparable aluminum articles. Three factors contribute to these high costs: (1) the learning costs involved in fabricating the parts, (2) material costs, and (3) design concepts. The

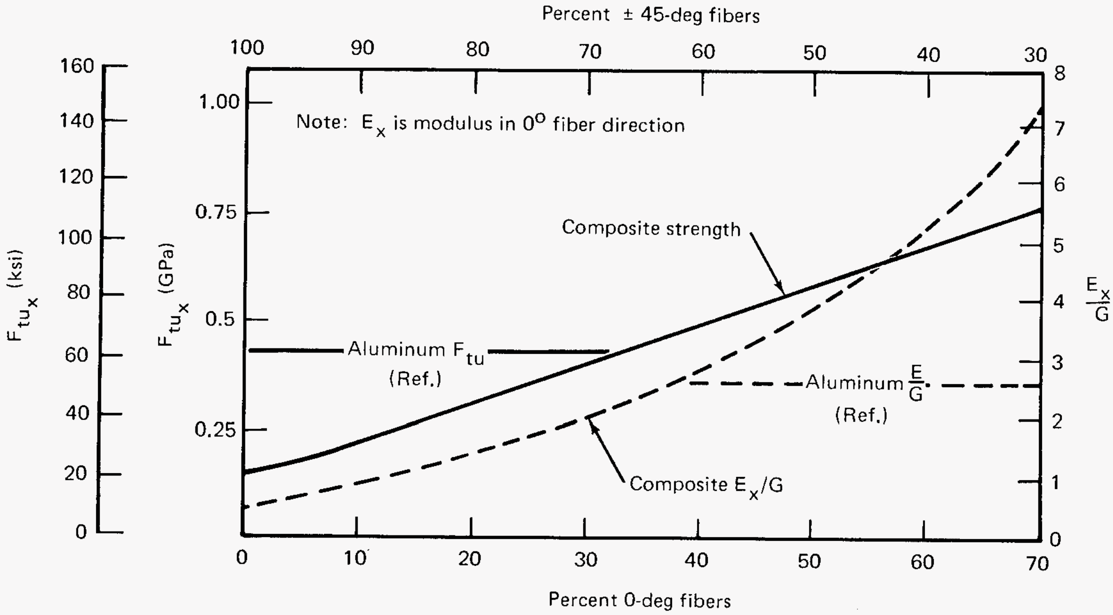


Figure 11. – Strength and stiffness variation of intermediate modulus graphite-epoxy laminates, $[0/\pm45]_c$.

learning costs are being reduced with every new program. The only apparent solution for material costs is a high production rate for the material. Design concepts become simplified through the progressive elimination of parts.

Reference 59 cites three production-type programs spanning a five-year period. The percentage of weight saved and the relative costs have become increasingly more attractive. Reference 132 states that fewer manufacturing hours were required for the first 50 composite parts than the first 50 metal parts. Even greater gains are possible when manufacturing considerations are included in the basic design process.

In summary, the producibility of composite structures is constantly being improved (e.g., the boat-hull fabrication technique for fuselage structure presented in reference 106). In fact, it is now possible to produce some composite structures competitively with current metal technology in a production environment (ref. 59).

2.2.9 Maintainability

Basically, a maintainable structure has many openings to permit easy access to components. However, a basic design objective for composite material is to minimize the number of openings, which cause stress concentrations. Thus, a conflict arises. The problem is complicated because the increased maintenance and repair costs of the optimum structure are partially offset by reduced manufacturing costs, since every stress concentration is expensive. The obvious solution is a new or different design concept for each particular situation as a means of satisfying both ends, or a compromise between maintenance and repair cost and an optimum structure.

Although the tradeoff between a maintainable structure and an optimum structure has to be made during the basic design of the structure, nearly all composite structures in the past have had to have the same configuration as the corresponding metal structure. Only now, when composite materials can be considered at the inception of the design, is the designer given any latitude to implement this tradeoff.

2.3 Fracture Control

Fracture control represents a set of policies and procedures intended to prevent structural failure from the initiation or propagation of cracks or crack-like defects during fabrication, testing, and service. Fracture control affects material selection, design concepts, fabrication, verification, inspection, and maintenance. The importance of fracture control is widely recognized for metallic materials in aircraft and space vehicles. Attention will now also be given to fracture control for advanced composite materials because they fracture in a brittle manner and are weakened by manufacturing defects and accidental damage.

Criteria for fracture control are given in reference 4 for space shuttle structure. Although the criteria were developed primarily for metallic materials, they are general and can be used as criteria for composite materials. However, analytical techniques for estimating the life and strength of composite materials containing defects (e.g., voids between plies or in the bondline of a joint) and accidental damage are not as well developed as for metallic materials. Thus, empirical techniques based on reliability theory are used for composite materials.

2.3.1 Control Plan

There is no generally accepted method for the fracture control of composite structure. However, the requirements for a fracture-control plan are given in reference 4 for a space-shuttle vehicle: the plan includes composite structures.

2.3.2 Analysis

The impact of fracture control is felt very early in composite design in that a suitable method for achieving the desired service life must be selected. Later, as the detail design develops, attention is given to those areas from which fracture is most likely to initiate, such as stress concentrations and manufacturing defects.

2.3.2.1 Service-Life Philosophy

Because of the relatively low efficiency of composite joints, the service-life philosophy must be carefully selected. The two methods for ensuring adequate structural integrity and service life are safe-life and fail-safe design.

A safe-life design is sized to allow for the expected initial flaws and degradation in residual strength during usage. In a deterministic design approach, this allowance is intended to ensure that the design service life is satisfied. In a reliability-based design, the allowance is intended to ensure that a specified reliability level is maintained throughout the service life.

A fail-safe design is configured so that after failure of any one structural element, the remaining structure has a residual strength adequate for continued operation until the ruptured element can be detected and repaired. In a deterministic design approach, a specified residual strength is required. In a reliability-based approach, a specified level of reliability is required.

A mixture of the two methods can be employed in the design of a given vehicle. The safe-life method is usually applied to statically determinate components or to components where multiple load paths are not practical. The fail-safe method is usually applied to components with indeterminate (multiple) load paths.

2.3.2.2 Stress Concentration Effects

The concept of an elastic stress concentration factor is applicable to composite laminates containing discontinuities 0.0254 m (1.0 in.) or larger in diameter (ref. 133). The magnitude of the stress concentration factor depends on the geometry and material anisotropy, and differs from the conventional value found in isotropic materials. The effect of orthotropy is demonstrated in figure 12 by the comparison of stress concentration factors along the cutout boundary for two boron-epoxy laminates and aluminum. Reference 94 presents closed-form solutions for the stress at various discontinuities in anisotropic plates. Since these closed-form solutions do not account for the finite boundaries encountered in practical design problems, alternate analytical methods have been employed.

Finite-element computer procedures developed specifically for this purpose are discussed in references 134 and 135. More recently, the integral-equation technique has been applied to this problem (ref. 136), and it offers the convenient option of solving for boundary stresses only. The complex variables approach has also been used successfully (ref. 137).

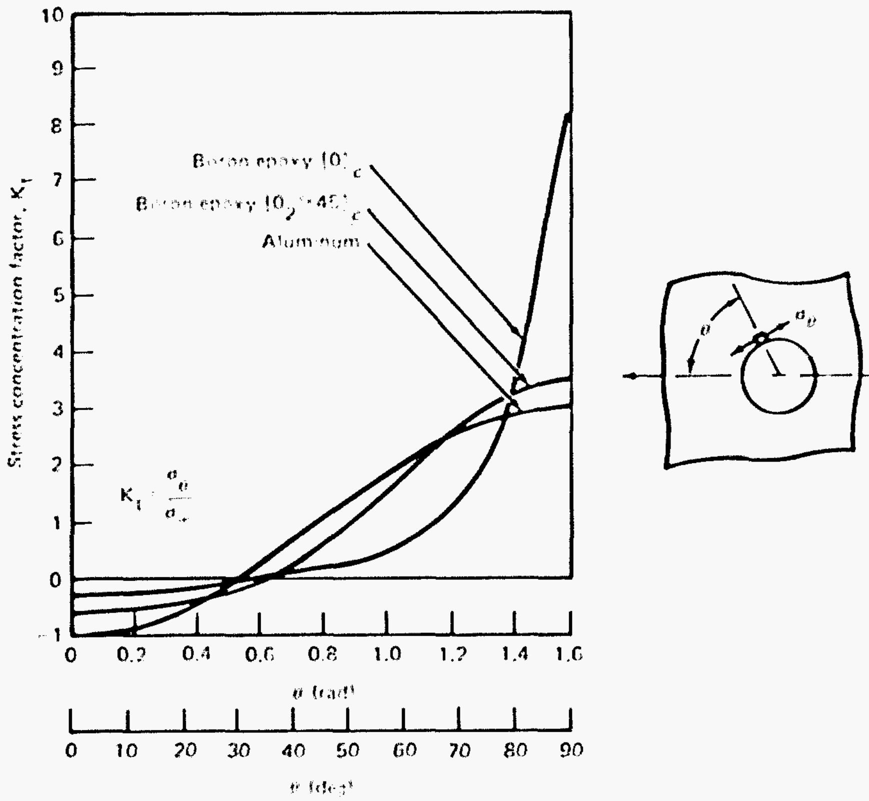


Figure 12. – Stress concentration factors along cutout boundary.

Boron and graphite filaments or fibers exhibit linear stress-strain behavior to failure. In fact, most composite laminates designed for structural applications exhibit the same behavior since the filaments are oriented to serve as the major load-carrying elements in the laminate. This linear laminate response permits use of linear versions of analytical procedures to predict failure strength when coupled with the proper failure theory. References 134, 135, and 138 show that laminates containing large discontinuity radii such as access holes, windows, and cutouts fail when the predicted composite strains on the cutout boundary exceed the composite strain capability. One such failure is illustrated in figure 13, which compares measured and predicted strains near an 0.0254-m (1.0-in.) circular hole in a graphite-epoxy laminate loaded in tension.

As the absolute size of the discontinuity is reduced, the concept of an elastic stress concentration factor no longer applies. As shown in figure 14, the strength of a tensile specimen containing a circular hole increased with decreasing hole diameter. Reference 133 proposes one method of modeling this behavior using linear-elastic fracture mechanics and assuming homogeneous anisotropic material behavior. Application of this method to a tensile specimen containing a central slit yields the lower curve in figure 14. An alternate approach considers both fiber and matrix behavior in modeling the fracture mechanism (ref. 139).

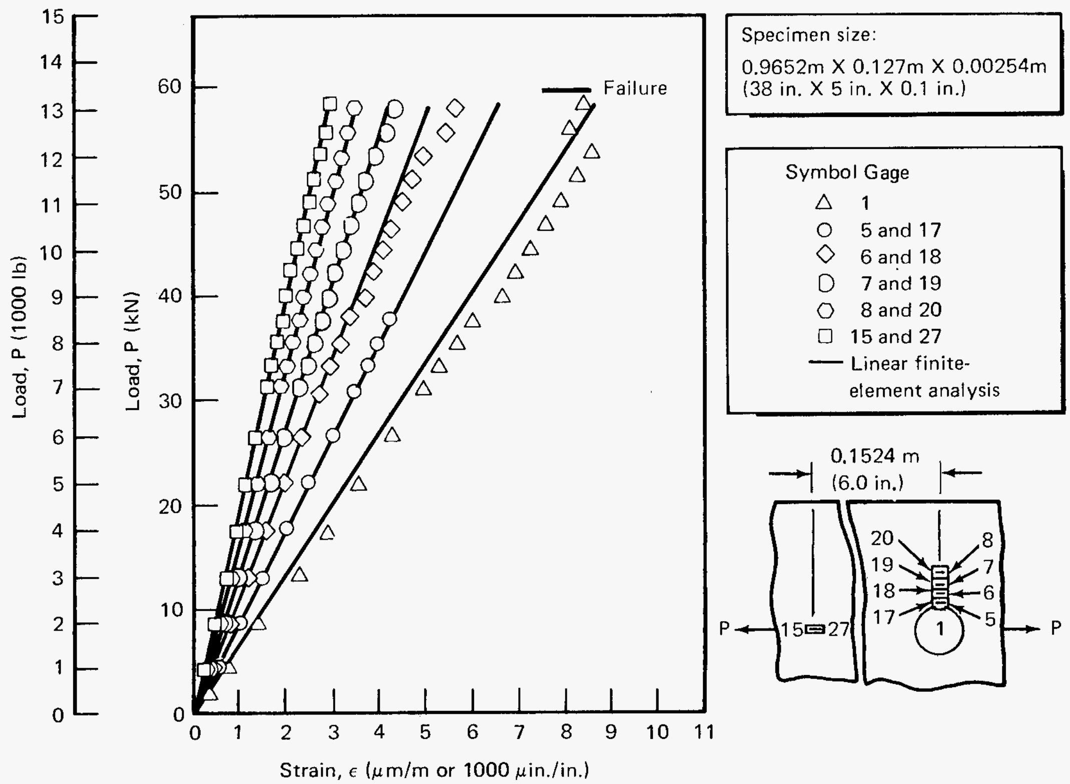


Figure 13. – Measured strains near 1.0-in. diameter cutout graphite-epoxy $[0/\pm45]_{2s}$

In summary, the fracture behavior of a composite laminate containing a discontinuity depends upon the following:

- Anisotropy of the laminate
- Effect of finite boundaries
- Size of the discontinuity
- Orientation of the discontinuity with respect to the laminate and load

Solutions to certain design problems related to fracture have been found by tailoring the properties of laminated materials. Certain laminates such as $[\pm45]_c$ loaded in tension do not exhibit linear stress-strain behavior. Laminates of this type have been employed around circular holes to reduce the stress concentration by permitting local yielding to occur (refs. 119 and 120). The low extensional modulus of the same

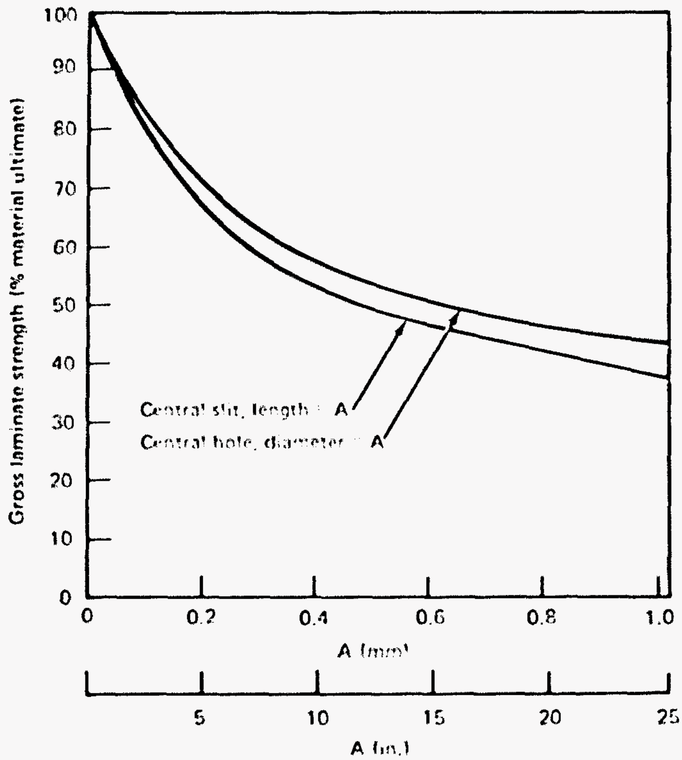


Figure 14. – Strength variation with flaw size in infinite plate under tension, $[0/\pm45/90]_{2s}$ boron-epoxy.

laminate $[\pm45]_c$ has been used to advantage in crack-arrestment tests: strips of the low-modulus laminate were placed at intervals in the primary load-carrying laminate (ref. 140). Low-modulus, high-strength strips (e.g., fiber glass-epoxy) have proved equally successful in reducing stress concentrations around holes (refs. 141 and 142).

Fatigue life is the point in time or the number of load cycles when an element fractures because the applied load exceeds the residual strength of the element. The stress concentrations in the element have a pronounced effect on the residual strength, and hence the life of the element. Figure 15 shows the effect of various types of stress concentrations such as open holes, bolted joints, and bonded joints on the fatigue life of a boron-epoxy laminate. The fatigue life of the basic laminate exceeds 10^7 load cycles even when cycled at limit stress. An open hole significantly reduces laminate static strength, but subsequent damage incurred during cyclic loading is less severe than in the basic laminate with no hole (ref. 133). Both bonded and bolted joints, however, exhibit significant loss in strength with cyclic loading similar to the loss of strength in notched metals (refs. 123, 143, and 144). Joints thus become the critical elements in the fatigue life of a composite.

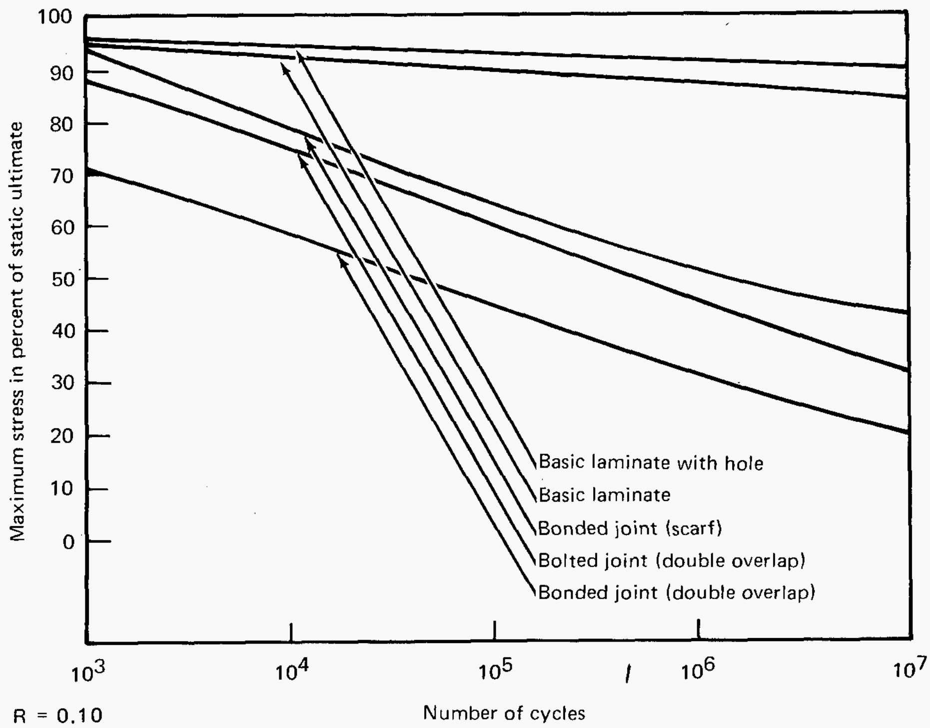


Figure 15. – Typical element fatigue behavior $[0_i/\pm45_j]_C$ boron-epoxy (ref. 10)

The most common manufacturing defect is a void between plies in a laminate or in the bondline of an adhesive joint. Voids of this type with a diameter greater than 0.00635 m (0.25 in.) are detectable with present inspection techniques. Should undetected voids grow under repeated loading, a degradation in residual strength may result. There are insufficient data on the fatigue behavior of composite laminates containing voids to quantify this phenomenon.

2.4 Design Verification

The verification of a composite structure usually involves three broad areas (1) material procurement and acceptance, (2) structural testing, and (3) quality assurance programs and methods.

2.4.1 Material Procurement and Acceptance

The procurement and acceptance of composite materials is a well-developed field, but the specifications that have been written are not uniform. A general trend is toward specifying a maximum coefficient of variation as well as the minimum value for the prescribed properties.

2.4.1.1 Procurement

The qualification of a vendor to supply a particular composite material system is usually governed by a procurement specification. Every company with an interest in composite material structures either has adopted an existing procurement specification or has written one. The major differences in these specifications lie in the way allowables are selected: some of the specifications require the use of minimum strengths from various tests, while others require statistical values for these parameters.

The following physical properties of boron-epoxy material (ref. 145) are representative of the properties contained in a qualification specification:

- Tack – degree of cure of the resin
- Flow characteristics of the resin during the cure cycle
- Volatile content
- Resin content
- Cured ply thickness
- Thermal expansion
- Outgassing
- $[0]_c$ flexure strength at RT, 406K (270°F), 450K (350°F), and 489K (420°F)
- $[90]_c$ flexure strength at RT, 406K (270°F), 450K (350°F), and 489K (420°F)
- Horizontal shear strength at RT, 406K (270°F), 450K (350°F), and 489K (420°F)
- Sandwich-beam face tension of $[0]_c$ and $[0/90]_c$ laminate

The types of tests and types and number of test specimens from which these properties may be obtained form a part of the specifications. Either minimum values or statistical failure values may be specified for the mechanical properties. Another condition that may be imposed is that the prospective material supplier must perform the tests on his product before he ships it to the buyer.

2.4.1.2 Acceptance

Acceptance tests are performed on all materials that have been previously qualified for procurement. These tests are performed on each shipment or batch, whichever is smaller, when the material is received for production. Boron-epoxy acceptance tests will be summarized here as an example (ref. 139).

Strength tests for $[0]_c$ flexure, $[90]_c$ flexure, and horizontal shear are required at RT and 450K (350°F). Flexure tests are used for the 0-deg-fiber-orientation strength tests; however, 0-deg tension tests may be substituted. The flexure tests are sometimes more representative of in-service loadings than tension tests.

Two types of specifications exist for these strength tests: one calls for a certain number of tests with the average test value to exceed a given minimum strength, and the other requires a sufficient number of tests to determine mean strength and statistical variation for the properties. The statistical variation requirement is imposed to exclude material with a large scatter in properties.

It is not customary to check all the physical properties as a part of acceptance testing. Although not a part of the specification, certain physical property tests (e.g., infrared spectral, thermal gravimetric, and thermal deformation analyses) are conducted by use of a "fingerprint" (spectrometric analysis) of the resin system; these tests alert the user to any change in the resin formulation in subsequent deliveries. Since boron-epoxy is commonly procured in the form of a monolayer of tape 0.0762 m (3.0 in.) wide, the width and position of the material on the paper backing are checked. Resin content and flow are also checked.

2.4.2 Structural Testing

Structural tests (strength, life, etc.) are usually the same for conventional metallic structures and composite structures. However, the tests are different for deterministic and reliability-based design approaches.

2.4.2.1 Strength and Deformation

Static tests of composite structures have been performed in the past as the final proof of design. The tests usually involve loading the structure to limit levels in several loading conditions followed by loading to failure in one of the critical conditions. References 6, 8, and 146 may be used as a guide for developing a test plan, test documentation, and the type of data required.

In general, the test methods and requirements that have been developed for metallic structures are applicable to composite structures. However, static strength distributions are usually not determined statistically for conventional structures since the safety factors are a result of design experience. In a reliability-based design approach for composite structures, these distributions are determined for appropriate element and subcomponent data.

2.4.2.2 Life

Fatigue life tests are commonly used to verify the design life of a composite structure. In the deterministic approach, the length of the test is equal to the design life multiplied by an appropriate fatigue scatter factor (2 to 4).

In the reliability approach, adequate residual strength is desired with a specified reliability goal at the end of one service life. Thus, a reliability approach requires knowledge of the statistical distribution of residual strength at the end of one life. This distribution can be estimated from a limited number of fatigue tests and with the appropriate element and subcomponent data.

2.4.2.3 Damage Tolerance

Damage-tolerance requirements have not received a great deal of attention in the composite demonstration structures fabricated to date. Damage tolerance becomes important, however, as usage moves from demonstration components to production flight hardware (refs. 147 and 148). Damage-tolerance levels form part of the specifications on a current RDT&E contract involving composite structures, and are demonstrated by analysis, or test, or both.

2.4.2.4 Dynamics

The effects of ground and flight dynamic loadings, flutter, divergence, vibration, sonic fatigue, and other aeroelastic instabilities are determined for composite structures in generally the same way as for metal structures. References 7 and 149 present procedures for determining these effects. Damping coefficients for the laminate vary with the laminate orientation over a wide range. Typical structures have shown damping coefficients of $g = 0.05$ to 0.06, depending upon frequency (ref. 131).

2.4.3 Quality Assurance

Quality assurance programs have been used for some time in the design of composite structures and the value of these programs has been demonstrated.

2.4.3.1 Destructive Evaluation

The destructive evaluation of composite parts involves the testing of test tabs and often of actual production parts. A test tab is a small coupon representative of a production part. Fabricated along with the production part, the test tab may be integral with the part or cured adjacent to the part that it represents. It thus represents the actual part with respect to material, tooling, cure cycle, and lay-up. Specimens are cut from the tab and tested to determine their physical and mechanical properties and thereby to verify correct processing. The production part's properties are assumed to be the same as those of the test tab.

2.4.3.2 In-Process Controls

In-process controls refer to all details and steps which must be observed and recorded to ensure that the material in the fabricated part will have the desired physical and mechanical properties. As a minimum, these properties are those previously specified and verified in the acceptance tests for the material (see Sec. 2.4.1.2).

In-process specifications are usually written for each type of composite material (e.g., refs. 145 and 150). These specifications cover all operations done on the material from the time it is removed from storage through the cure cycle of the part.

2.4.3.3 Acceptance Testing

Parts are accepted for use in the structure on the basis of inspections and tests. The entire part is first inspected for such defects as gaps and wrinkles in the material, foreign objects, contour discrepancies, and dimensional deviations from the drawings.

A thickness survey of the part is conducted next. The total thickness at any point is divided by the number of plies at that point to arrive at a per-ply thickness survey over the part. This survey indicates resin-rich or resin-poor areas. Ultrasonic inspection is usually used to check for debond areas.

Hardness measurements are sometimes taken along the outer edge of a part in the trim areas. These measurements indicate a properly cured part by comparison with reference hardness values. Finally, the test tabs are cut into specimens and tested to determine the *in situ* properties of the materials.

The proof-testing of metallic pressure vessels is covered in references 3 and 4. The testing procedures of these documents generally apply to pressure vessels made of composite materials. In addition to pressure vessels, flight-critical structure may be proof- or acceptance-tested to ascertain whether the structure contains critical flaws.

Environmental conditions are usually included in the proof tests if they can significantly affect the strength of the structure. Further data are needed to assess the desirability of proof-testing versus more rigorous nondestructive inspections.

2.4.3.4 In-Service Inspection

In-service inspections are usually visual inspections of the entire structure or part for scratches, dents, abrasions, penetrations, ruptures, delaminations, and other damage. The extent of the damage may be indicated by broken, exposed, or disintegrated fibers. If the edges of the part are visible, they are checked for delaminations. Nondestructive inspection techniques are used to locate internal voids or debonds. Flight-critical parts of the structure usually undergo in-service inspections.

2.4.3.5 Repair

The two most general repair techniques for local damage of laminated composite structures are (1) patching a hole or damaged area, and (2) injecting resin into a debonded area. It is generally accepted that almost any local damage can be repaired by one of these methods.

One method of patching a damaged skin area first involves removal of the flawed material, which may be done by machining a step-wise or beveled edge for an inlay or overlap repair. A patch of titanium is then inlaid and bonded in the hole. Repair kits are available for this type of repair (ref. 151). However, this method is limited to minor damage in thin skins.

More extensive damage can be repaired with a similar technique in which the patch is of the same material and ply orientation used in the damaged area. The patch can be cured separately and bonded into the hole or it can be laid up wet and cured in place. Field repair kits have not been developed.

Delaminations are usually repaired by injecting resin into the void or by removal of plies and inlaying a patch. After the delaminated area has been defined, small holes [1.016 mm (0.040 in.) in diameter] are drilled into the surface down to the void depth. Resin is injected into the void, and then cured under heat and pressure to complete the repair (ref. 152).

3. CRITERIA

The basic constituents of the composite material shall be selected on the basis of their mechanical and physical properties. The design of the material system shall account for these properties and other appropriate parameters. The design of the composite structure shall be conducted at either the lamina or laminate level. Design data for the lamina or laminate shall be determined statistically. The design shall account for the design conditions, reliability (for a reliability-based design), design factors (for a deterministic design), scaling from the material level into the component level, loads, strength, stiffness, producibility, and maintainability. Whenever feasible, a fracture control plan shall be established to ensure that unacceptable structural fractures will not occur during the service life. All stress concentrations shall be accounted for. Tests and analyses shall be performed to verify the design. The basic composite materials shall be qualified through accepted procurement, process, and inspection procedures.

Strength, life, dynamic response, and damage-tolerance levels shall be demonstrated by analysis, or test, or both. A quality assurance program shall be specified for the structure during and after fabrication.

3.1 Materials

3.1.1 Material System Design

3.1.1.1 Basic Constituents

The mechanical and physical properties of the fiber and the matrix shall be determined in the design of a material system for a particular application.

3.1.1.2 Systems

In any composite material system, the following factors shall be evaluated: (1) fiber diameter, (2) fiber spacing, (3) fiber surfaces, (4) fiber-to-matrix volume ratio, (5) mechanical and physical properties, (6) volatiles content, (7) carrier systems, (8) void content, and (9) processing parameters.

3.1.2 Material Design Levels

One of the following two basic levels of design shall be used for composite structures: (1) lamina level, with ply orientation and thickness as design variables, or (2) laminate level, with thickness only as the design variable.

For design at the lamina level, test data shall establish the following:

- The mean and variance of the lamina strength over the range of environmental conditions required by the design situation

- Translation of lamina properties into laminate properties

For design at the laminate level, test data shall establish the mean and variance of the laminate strength over the range of environmental conditions required by the design application.

Controls shall be established to ensure that acceptable values of the material properties are retained throughout the production cycle.

3.1.3 Material Characterization

3.1.3.1 Statistical Design Data

The design data for a material system shall account for the type of application, the anticipated service life, and the operating environment for the composite structure. The following shall be developed statistically:

- Material static and residual strength data

- Material life data

3.1.3.2 Mechanical and Physical Properties

The following mechanical properties of lamina or laminate material shall be determined for use in design: (1) longitudinal modulus, (2) transverse modulus, (3) in-plane shear modulus, (4) in-plane Poisson's ratio, and (5) tension, compression, and shear strength associated with fiber and transverse directions. Other mechanical properties such as fracture and creep characteristics of the material system shall be determined in conjunction with the anticipated service life of the composite structure (including the operating environment) for either the lamina or laminate design level (Sec. 3.1.2).

The following physical properties shall also be determined for the lamina or laminate: (1) material density, and (2) coefficients of thermal expansion in the longitudinal and transverse directions. Additional physical properties shall be determined as required.

3.2 Design

3.2.1 Management

For a deterministic design approach, appropriate design factors shall be established.

For a reliability-based design approach, at least the following shall be established:

- Fleet reliability goal
- Fleet size
- Reliability goals down to the component level of design
- Appropriate damage-tolerance requirements
- Feasible plan for implementing the reliability program

3.2.2 Design Conditions

The limit design conditions for the composite structure shall be defined by the planned operational usage of the structure. For a deterministic design approach, the ultimate load conditions shall be defined by multiplying the corresponding limit loads by an ultimate design factor, and the fatigue design spectrum shall be based on expected load statistics.

For a reliability-based design approach, the limit load condition shall remain as defined above; an overload condition shall be defined, together with the fatigue design spectrum from a random load history which preserves the expected load statistics of the composite cumulative exceedance data. The overload conditions shall be defined by conditions not expected to occur in the operational use of the structure.

The thermal and chemical environment expected during the service life shall be included in the fatigue requirements as appropriate.

3.2.3 Design Factors

For a deterministic design approach, the design factors for metallic structural design shall serve as the initial design factors for composite structure. These factors shall be appropriately modified as experience dictates.

For a reliability-based design approach, the design procedure shall account for the effect of structural strength scatter and fleet size and shall ensure the reliability goals for both the overload condition and the required service life (fatigue) condition.

3.2.4 Environment

The effects on composite structure of all environmental conditions that are not included in the fatigue life tests or static tests shall be established experimentally, or analytically, or both. Environmental protection shall be provided where necessary.

3.2.5 Scale Effects

The scaling process shall account for the translation of basic material design data into full-scale component design data.

3.2.6 Reliability

For a reliability-based design approach, a reliability goal shall be established for individual components on the basis of the fleet reliability goal, which includes both fatigue and overload.

3.2.7 Analysis

3.2.7.1 Internal Load Determination

Analysis of the internal load distribution for a component shall account for all applied loads, the structural arrangement, material lay-ups, load paths, elastic response requirements, and localized responses.

3.2.7.3 Laminates

Composite laminates shall be designed to meet all strength and stiffness requirements. Full advantage shall be taken of the possible strengths resulting from laminate orthotropic or anisotropic properties. The influence of stacking sequence, laminate asymmetry, Poisson's ratio as related to free-edge effects, and transverse stresses resulting from out-of-plane loads shall be minimized unless shown to be beneficial.

3.2.7.3 Panels

For a deterministic design approach, panels shall not fail at ultimate load, nor shall any deformation resulting from limit loads produce changes in stiffness or load distribution that degrade the integrity of the panel or any other system.

For a reliability-based design approach, panels shall be designed to meet their apportioned reliability goals for fatigue life and overload capacity.

The panel design and analysis shall account for all combinations of mechanical, thermal, and residual stress loading resulting from the expected service conditions. The effect of environmental conditions upon the material properties shall be accounted for in the design. The panel analysis and design shall include at least the following:

- Cutouts
- Local failure modes
- General instability
- Delaminations resulting from combined loading effects, including thermal loadings

3.2.7.4 Shells

For a deterministic design approach, shells (with or without stiffening) shall not fail at ultimate load. At limit load, deformations shall not degrade the integrity of the shell itself or of any other system.

For a reliability-based design approach, shells shall be designed to meet their apportioned reliability goals for fatigue life and overload capacity.

Cutouts, elastic end supports, and other special problems such as nonuniform stiffnesses, variation of load with time, and effect of initial imperfections shall be accounted for.

3.2.7.5 Joints

The assembly stresses of components and subcomponents shall be determined through analysis, or test, or both. Joints shall be treated as a separate material system which must be characterized statistically. At least the following shall be accounted for as part of the overall design approach:

- Loads that must be transferred
- Available area for the transfer
- Geometry of the members to be joined

- Stress distribution in all regions of load transfer
- Service environment to be experienced
- Service life
- Reliability of the joint (for the reliability-based design approach)
- Thermal strains
- Assembly stresses

The applicability of safe-life and fail-safe design concepts to joint design shall be determined. These approaches shall be incorporated as appropriate.

3.2.7.6 Component Design for Strength and Stiffness

The component static strength and stiffness shall be sufficient to sustain the limit loads and pressures in the expected operating environment throughout the service life without experiencing detrimental deformation for a deterministic design approach or without degrading the reliability below the specified level for a reliability-based design. In addition, the component stiffness shall be such that the component is free from aeroelastic instability and free from deformations that will degrade stability and control below specified limits throughout the service life.

For a deterministic design approach, a structure shall withstand ultimate loads under all anticipated environmental conditions without experiencing rupture or collapse. For a reliability-based design approach, the structure shall withstand the overload condition under all anticipated environmental conditions with a specified residual strength throughout the service life.

3.2.8 Producibility

The structural design shall employ only proven processes and procedures for production manufacturing.

3.2.9 Maintainability

The composite structural design shall permit the structure to be maintained within specified limits. The design shall allow accessibility for inspection, repair, and maintenance of critical components.

3.3 Fracture Control

3.3.1 Control Plan

A fracture control plan shall be developed along the lines of reference 4 and shall include provisions for at least the following:

- Identification of components selected for fracture control on the basis of their criticality
- Establishment of a data bank of fracture information
- Maintenance of a continuing quality assurance activity directed toward identifying and reporting conditions that could affect the fracture behavior of structural components

3.3.2 Analysis

3.3.2.1 Service-Life Philosophy

Each fracture-controlled component shall be evaluated to determine whether a safe-life or fail-safe design or a combined approach is more appropriate.

3.3.2.2 Stress Concentration Effects

Effects of stress concentrations on the fracture behavior of the controlled components resulting from design constraints, manufacturing defects, and repairable in-service damage shall not reduce the required life of a composite structure (deterministic design) or reduce the specified reliability level (reliability-based design). The fracture toughness of a particular laminate shall be accounted for.

3.4 Design Verification

3.4.1 Material Procurement and Acceptance

3.4.1.1 Procurement

Procurement procedures shall define the means of qualifying a material supplier for a particular material. The specifications shall contain either minimum or mean statistical values for physical and mechanical material parameters and a specified coefficient of variation. The testing procedures required to determine these properties shall also be specified and implemented throughout the production cycle.

3.4.1.2 Acceptance

Incoming acceptance tests shall be prescribed. These tests shall be performed on each shipment or batch, whichever is smaller, of material received from a previously qualified supplier. These tests shall include dimensional checks and tests for the physical and mechanical properties of the material.

3.4.2 Structural Testing

3.4.2.1 Strength and Deformation

It shall be demonstrated by tests that the structure does not deform under design limit loads in a manner which adversely affects vehicle performance. It shall be demonstrated by tests that the structure does not fail under design ultimate loads (deterministic design). For a reliability-based design, it shall be demonstrated by test that the structure retains a specified level of residual strength in the overload condition.

3.4.2.2 Life

For a deterministic design, it shall be demonstrated by tests that the structure exhibits a fatigue life equal to the design life multiplied by a given scatter factor.

For a reliability based design, it shall be demonstrated by tests that the structure exhibits a specified minimum level of reliability throughout the service life.

3.4.2.3 Damage Tolerance

Damage-tolerance levels shall be specified and demonstrated by structural tests, or analysis, or both.

3.4.2.4 Dynamics

The ability of the structure to withstand all dynamic loading conditions anticipated throughout its service life shall be demonstrated by appropriate tests, or analysis, or both.

3.4.3 Quality Assurance

3.4.3.1 Destructive Evaluation

Test tabs or representative parts of flight-critical components shall be destructively tested. Test specifications shall be established.

3.4.3.2 In-Process Controls

In-process controls and tests shall be specified to ensure that the material being used has not been damaged and that it is fabricated correctly.

3.4.3.3 Acceptance Testing

Inspection procedures and accept-reject criteria shall be specified for the nondestructive inspection of all parts fabricated with composite materials. This inspection shall be capable of detecting the defects which could diminish the structural capability, flightworthiness, fatigue life, or environmental resistance of the structure. The quality of flight-critical structure shall be demonstrated by proof-testing or by an accepted end-of-the-line nondestructive inspection plan, as appropriate.

3.4.3.4 In-Service Inspection

The design of composite structure shall include adequate provisions for in-service inspection of flight-critical parts or areas of the composite structure.

3.4.3.5 Repair

The design of composite structures shall permit repair without degrading flightworthiness.

4. RECOMMENDED PRACTICES

4.1 Materials

4.1.1 Material System Design

4.1.1.1 Basic Constituents

Fibers should be selected primarily on the basis of strength, strength variability, fabricability, modulus, density, and thermal expansion. Matrices should be selected on the basis of environmental and producibility requirements. Particular attention should be paid to the response of the matrix material to temperature, moisture, corrosion, and the space environment.

4.1.1.2 Systems

The procedure of transforming fibers and matrices into specific material systems should be considered as an engineering art. Thus, the use of mature, fully developed systems is strongly recommended. Only qualitative judgments should be made regarding the performance of a new material system, since little is known about fiber-matrix interaction.

Acceptance tests and photomicrographs should be used to evaluate the factors listed in Section 3.1.1.2 except for processing parameters, which should be evaluated on the basis of experience, since such variables as tack and shelf life can only be assessed qualitatively.

The design of the composite system should account for the transfer of fiber and matrix properties to lamina properties. As a general rule, only those systems with a minimum strain-to-failure capability at room temperature of 7 mm/m (7000 μin./in.) in the fiber direction and 4 mm/m (4000 μin./in.) transverse to the fibers should be used. If higher-modulus systems with correspondingly lower strain capabilities are used, they will be subject to accidental damage.

4.1.2 Material Design Levels

The design procedure, whether performed at the lamina or laminate level, should begin with definition of a strength criterion. Since the particular failure criterion selected for laminate composites is currently immaterial with respect to combined loadings, it is recommended that the failure criterion be based upon lamina-failure strains or where feasible, the laminate *in situ* lamina-failure strains. This recommendation is based solely on the fact that the stress-strain response of the laminate is a sum of the responses of the individual plies. Consequently, laminate failure is precipitated by lamina failure, and design ultimate stress for the laminate is determined by first-ply failure (the 90-deg lamina). This practice results in some conservatism, as shown in figure 16.

The response of a selected laminate to the expected loads and environment should be determined. Residual strength and life data should be generated for the laminate under one random spectrum, such as a Gaussian distribution, and for several root-mean-square stress levels.

Laminate residual-strength data should be used in conjunction with the failure criterion to define an interaction diagram as a function of life. Diagrams such as these should be used for initial selection of material systems and laminate orientations.

Fatigue-life data for the candidate laminate should be considered next. If the candidate laminate does not exhibit the required life characteristics, then another laminate orientation or material system should be selected. This process should be continued until a material system and laminate orientation are found which meet the preliminary design requirements. The design process should then proceed as described in Section 4.2.5.2.

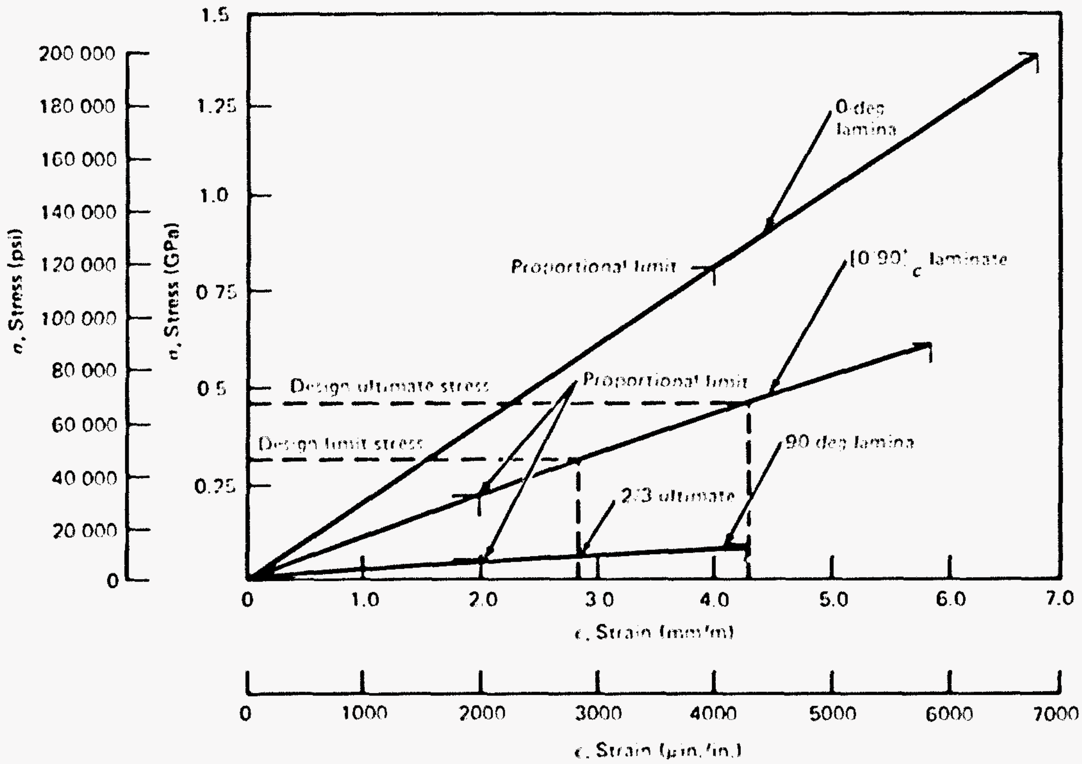


Figure 16. – Relationship between lamina and laminate strength for typical boron-epoxy $[0/90]_c$ laminate.

4.1.3 Material Characterization

4.1.3.1 Statistical Design Data

It has been demonstrated that statistical lamina data are transferrable to cross-plied laminates if the failure mode (fiber or matrix) remains constant (ref. 153). Thus, if one can predict the mean strength for the laminate (or the scale parameter β of a Weibull distribution), then the shape of the laminate strength distribution (given by the coefficient of variation or the shape parameter α of a Weibull distribution) can be assumed to be the same as for the lamina. This concept is illustrated in figures 17 and 18 for the fiber-controlled and matrix-controlled tension failure modes, respectively. It has further been demonstrated that these statistics are valid at temperatures other than room temperature (refs. 153 to 155). However, restrictions must be placed on the temperature range because of the existence of transition temperatures in the matrix material. This is shown in figure 19 in the change in the shape parameter above 450K (350°F). It is established that the epoxy resins exhibit a heat distortion (glass-transition) near this temperature.

Because of this heat distortion temperature, statistical design data should be generated not only at room temperature but also at the temperature range of interest. Mean

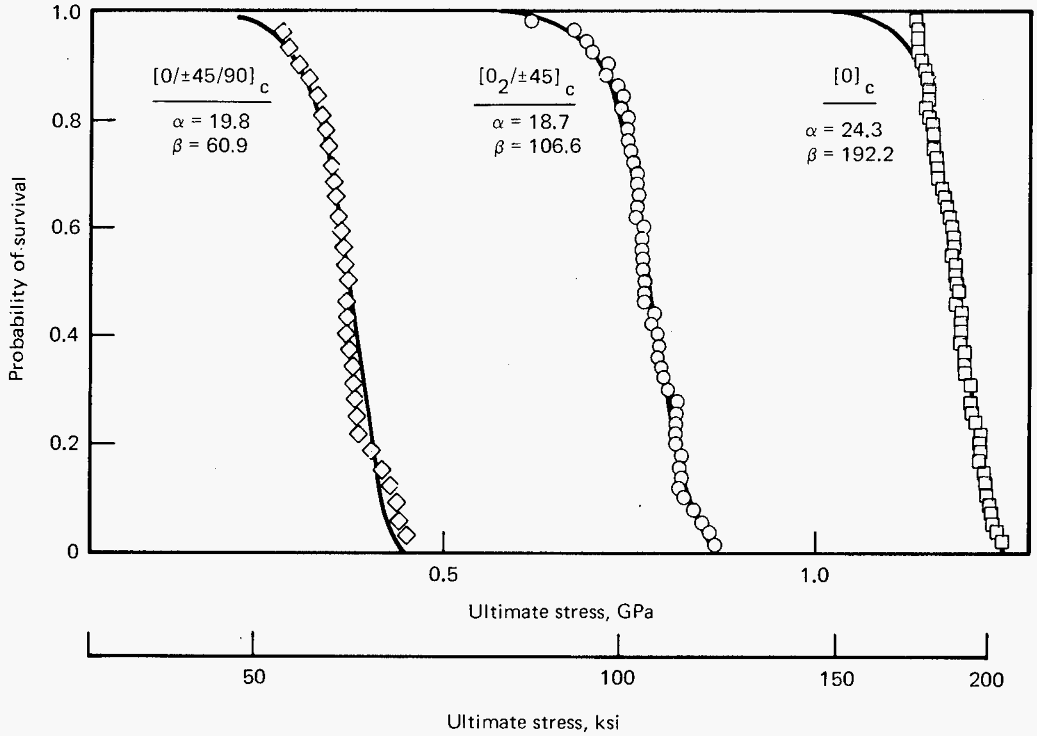


Figure 17. – Fiber-controlled lamina-laminate coupon comparison.

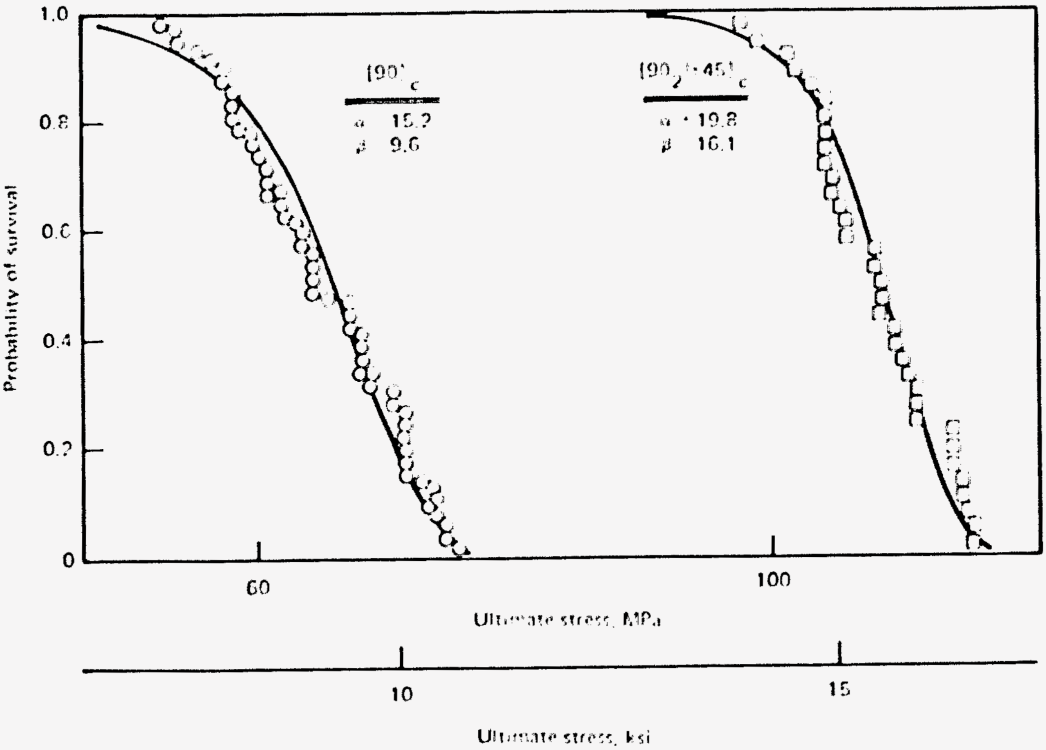


Figure 18. – Matrix-controlled lamina-laminate coupon comparison.

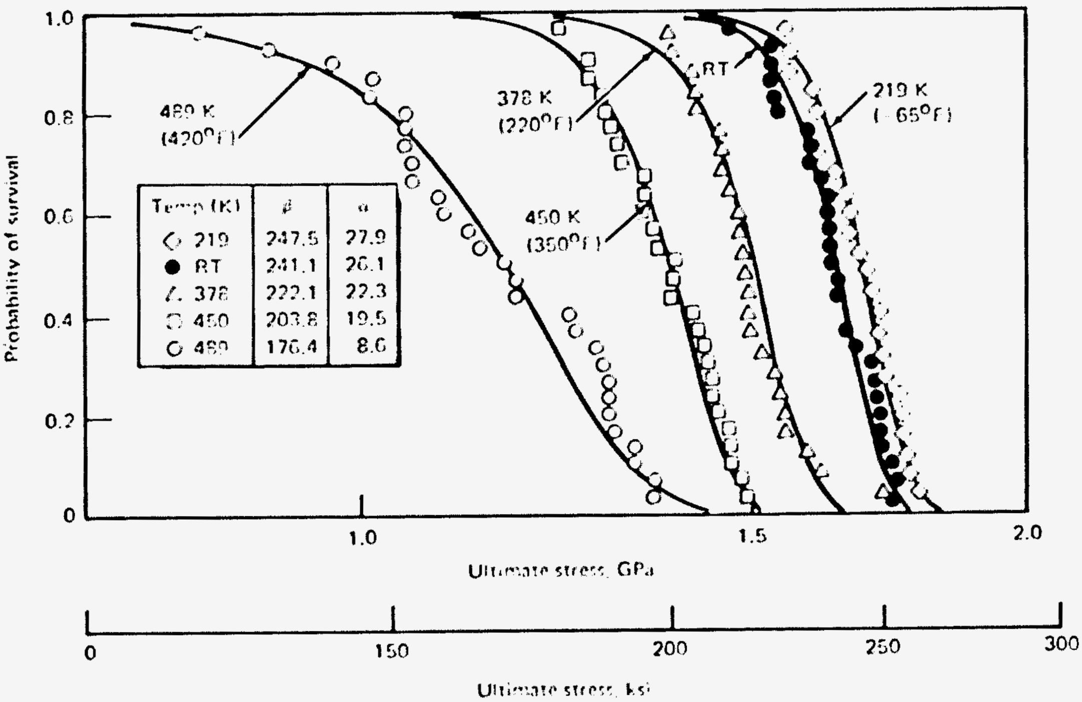


Figure 19. – Effect of temperature on $[0]_c$ boron-epoxy.

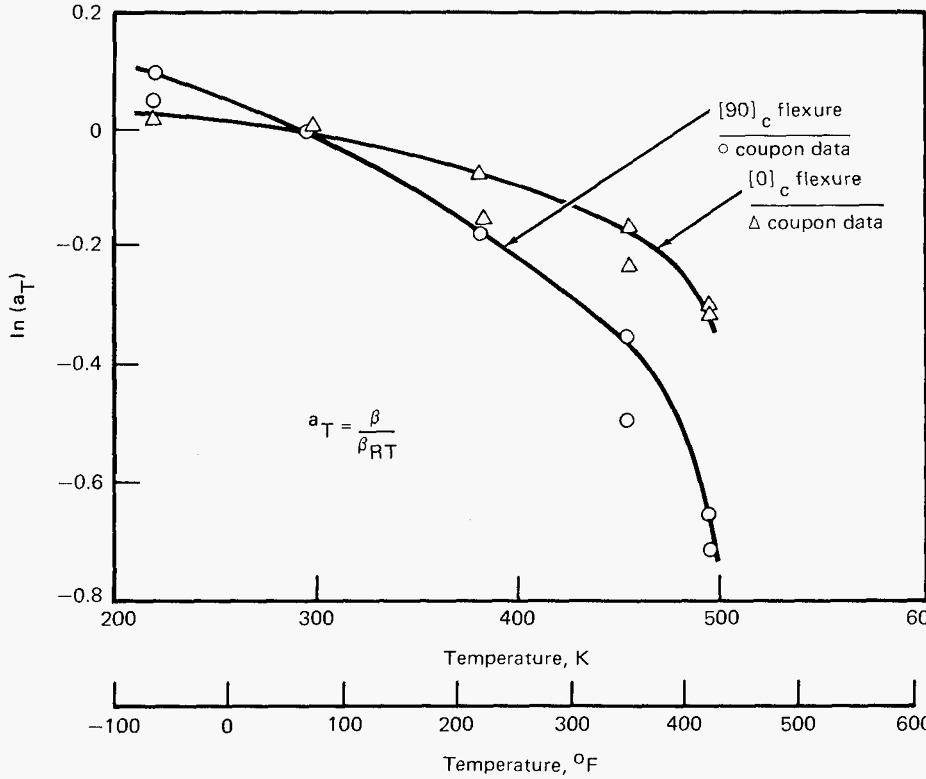


Figure 20. – Strength retention of boron-epoxy at various temperatures.

strength retention should be plotted as the natural logarithm of a_T (the ratio of β at a specific temperature to β at room temperature) vs. temperature, as shown in figure 20. Mean strength data presented in this form provide information on the location of heat-distortion zones and forewarn the designer of variations in statistical behavior. These curves should be generated for the various failure modes, such as tension, compression, and interlaminar shear. Similar retention curves of a_T vs. temperature should be constructed to check the heat-distortion temperature.

The requirement for statistically significant data suggests testing large numbers of specimens. However, as long as the failure mode remains constant, as in figures 17 to 19, the shape parameter α remains constant. If the scale parameter β is properly shifted, data from different laminates (fig. 17) can be pooled together (ref. 156) with data from various temperatures (fig. 19). Therefore, increased confidence can be obtained in the shape parameter α without a large-scale test program. Data should be pooled wherever possible.

Statistically significant fatigue data should also be generated for the laminate. Wherever possible, all specimens should be tested to laminate rupture. However, the time and cost of fatigue testing at low root-mean-square stress levels may become prohibitive.

Therefore, statistical techniques derived from the theory of extremal statistics should be applied. For example, reference 157 discusses the concept of least-of-N testing to generate fatigue data.

4.1.3.2 Mechanical and Physical Properties

Mechanical properties of both lamina and laminate (e.g., elastic moduli, Poisson's ratio, and thermal expansion coefficients) should be determined on an average basis and as a function of temperature. Although these properties are normally established under static conditions, recent work in fracture mechanics suggests that such properties may change with age. Therefore, wherever possible, mechanical and physical properties should be evaluated as a function of time (ref. 158). For structures subject to various loading rates, the influence of strain rates on structural properties should be determined.

4.2 Design

4.2.1 Management

In the deterministic design approach, design factors for both static and fatigue loadings should be established as recommended in Section 4.2.3.

In a reliability design approach, the damage-tolerance requirements and the attendant reliability goals should be established first. For fail-safe structure, the reliability goal reflects inspection intervals and frequency of repair. For safe-life structure, the reliability goal reflects inspection intervals and probability of survival. Thus, the extent of maintenance or degree of risk involved for the particular structure determines the reliability goal. It should be established on a fleet basis as a fixed probability of survival (e.g., a reliability of 0.999 implies that only one structure out of 1000 will not meet performance requirements during its life or maintenance intervals). The plan for implementing the reliability program should be the responsibility of the contractor, but it should be approved by the contracting agency.

4.2.2 Design Conditions

The limit design condition for composite structures should be defined in the same manner as it has been in the past for metallic structure (i.e., the expected operational extreme).

In a deterministic design approach, the ultimate load condition should be determined by multiplying limit loads by the appropriate ultimate design factor (ref. 56). For a reliability-based design approach, the overload condition should be defined from load-exceedance data. Thus, a load-exceedance curve is used to choose a load level that will be exceeded a specified number of times in the vehicle life. The fleet size should enter into the choice of the overload level (e.g., the overload may be set at a level which would be exceeded only one time in the fleet life). Figure 21 shows a load level which would be exceeded only one time in a fleet of 1000 structures.

A fatigue spectrum that represents the usage of the vehicle during its service life should be a design condition. Cycle-by-cycle, random-load-generation techniques should be used to simulate the variation in loads with time on a multisegment and multimission basis. The cumulative exceedance statistics should be preserved in the random simulation. The random fatigue spectra should be used in the generation of the data base as well as in the component fatigue tests.

4.2.3 Design Factors

The design factors in current use with most composite structures are the same as those used with analogous metallic structure. Reference 56 presents the deterministic design factors which should be used for the design of manned space vehicles unless otherwise specified. Fatigue design factors of 2.0 to 4.0 should be used for composite structure. Design factors are not used in the reliability-based design procedure.

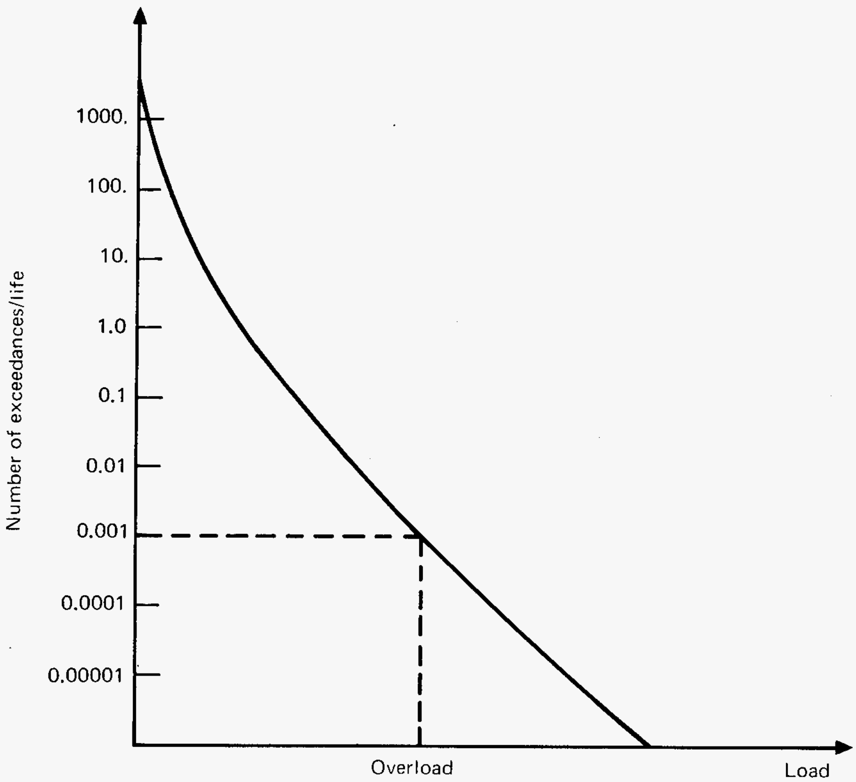


Figure 21. – Load level vs. exceedances per life.

4.2.4 Environment

Where applicable, thermal environments should be included in fatigue tests. Chemical environments (e.g., rain, salt water, humidity, and fuel) should be included in fatigue tests where practical or where no correlating data exist (ref. 159). Other environments such as radiation and lightning as well as the chemical environments not included in the fatigue tests should be evaluated as appropriate.

Protection against environmental effects should include the following, as appropriate:

- Plastic surfaces exposed to sunlight (UV) should be painted.
- Faying surfaces between dissimilar materials should contain nonconductive films or separators.
- Resin systems should be selected for minimum susceptibility to water vapor and then cured at or above 450K (350°F).
- For space application, resin systems should be selected for minimum outgassing.
- Plastic surfaces exposed to severe abrasion (e.g., leading edges) should be protected by metal films. In addition, a 3-mil layer of fiber glass should be an integral part of all external surfaces of the laminate.
- Lightning protection should be provided and such systems should be grounded to static discharge systems (ref. 147).
- Matrix systems should be selected with adequate thermal stability for the intended application. Care should be exercised when matrix-governed failure modes (particularly interlaminar) are critical at or near the maximum temperature limit of the material system (ref. 160).

4.2.5 Scale Effect

Material data should be translated from one level to another with due consideration of scale effects. In the deterministic approach, this translation involves updating allowables and imposing or increasing design factors. In the reliability-based approach, the translation involves modification of statistical parameters such as the shape parameter α of a failure distribution.

The first design level is the basis for material selection. This data base should contain initial estimates of mean strength and variance of both the initial strength and the residual strength at the completion of some number of lives. The data base should also account for the design parameters required for laminate design in a particular application.

The basic structural elements form the second level of design. Fatigue testing of these elements should be conducted with a simulated random-load history and in an appropriate environment for both design approaches. In addition, multiple small-scale testing should be conducted to determine the statistical data needed for the reliability-based process.

In a deterministic approach, subcomponents should be tested to validate or modify allowables and design factors. In a reliability approach, subcomponents should be tested to determine their residual strength and life distributions. The statistics of these tests should then be used to reevaluate the design, assess scaling effects, and predict the reliability of full-scale components. These tests should also be conducted with the random-load history and simulated environmental conditions.

In a deterministic approach, full-scale components should be tested to validate or modify allowables and design factors. In a reliability approach, a limited number of full-scale components should be tested to determine their residual-strength distributions and life statistics. These component statistics should then be used to predict the reliability and characteristics of the total composite structure.

4.2.6 Reliability

The reliability aspects of a structure do not explicitly enter the deterministic design process. Thus, there is no recommended practice for ensuring reliability.

Reference 68 provides "... common general requirements for all NASA programs to design reliability into aeronautical and space systems, and prevent degradation of the reliability of the design through the succeeding steps from fabrication to end use." Since such procedures have heretofore not been widely used for composite structures, Appendix A presents a proposed rational procedure for designing composite structures on the basis of their fatigue and chance-overload characteristics and their given reliabilities. Similar procedures are currently being used by various organizations in the aerospace industry.

4.2.7 Analysis

4.2.7.1 Internal Load Determination

The analysis of individual detailed parts of a composite component begins with the definition of the loads applied to the part. The determination of internal loads should be accomplished by a proper analytical modeling of the entire component. The complexity of these structures and the associated methods of analysis have led to an increasing use of numerical methods, with special emphasis upon the method of finite elements. The finite-element procedures, which have been experimentally verified, are therefore recommended when the classical approaches are nonexistent or invalid.

4.2.7.2 Laminates

The design of composite laminates must satisfy stability, stress, and dynamic requirements. Computer techniques should be used to arrive at an optimum orientation family. Otherwise, the number of possible material combinations and ply orientations is unmanageable. The point-to-point variation in the resulting laminate can be simplified to arrive at a representative orientation family. Then, a series of interaction or allowable envelopes should be developed with computer procedures such as SQ5 (ref. 85).

The final laminate design should also account for manufacturing constraints such as tape width and thickness and termination angles, especially when automated tape-laying equipment is used. The symmetry of the laminate about its midplane should be maintained within practical limitations. This is not always possible, but the effect of the nonsymmetry can be reduced by limiting the unsymmetrical plies to those nearest the midplane.

The free-edge effects should be considered in arriving at an optimum stacking sequence. The interlaminar normal stresses should be compressive rather than tensile in the free-edge zone to achieve optimum protection against delamination. For load-reversal cases, this will not be possible. Similarly, Poisson's ratio effects should be determined by utilization of automated procedures such as SQ5, and by considering strain compatibility between the composite laminate and laterally attached members when selecting an orientation family or combination within the family. Dispersing plies of various orientation through the thickness minimizes these effects.

Minimum gage for general-purpose and shear-only laminates should be six and four plies, respectively. The total number of plies may be divided equally between the two faces for sandwich construction. In all cases except pure shear, the laminate should contain at least three ply orientations — for example $[0 \pm 45]_c$.

4.2.7.3 Panels

The analysis of structural panels for buckling and crippling strengths (ref. 2) should follow the determination of the internal loads as described in Section 4.2.6.1. Finite-element procedures applied to the overall structural component define the deformations of the individual panels and the loads applied to these panels. The material properties should be determined by the equations in reference 82 or automated procedures such as SQ5 (ref. 85). The analyst should remember that any changes in panel stiffness due to changes in ply proportions or ply orientations must be reflected in the overall structural analysis.

The structural analysis phase of the design effort should include both stability and stress analyses for the various load conditions and environments. In many panels, stability considerations will determine the major characteristics of the panel. Stability analyses of composite plates should be conducted with orthotropic-plate theory (ref. 94). Methods such as those in reference 102 should be used to determine the static deflection, stability, and dynamic response of anisotropic plates.

Limited data on postbuckling response (refs. 104 and 105) indicate that caution should be exercised and substantiating tests included in the overall approach. The importance of the panel edges, joints, attachments, and cutouts cannot be overemphasized. The problems associated with mechanical joints are considered in Section 4.2.7.5.

4.2.7.4 Shells

For the analysis of axially compressed composite cylindrical shells, the method in references 115 and 116 should be followed. Although this method is intended for an existing shell on which imperfection amplitude data have been measured, it may be used with estimates of the expected imperfections to begin the analysis.

Until imperfection sensitivity analyses are performed on other shell configurations and loadings, the procedures outlined in references 1 and 91 or equivalent procedures should be followed. To treat configurations other than stiffened and unstiffened shells, one should use a general computer program such as NASTRAN (ref. 83) and BOSOR (ref. 109) to perform classical analyses to guide the design.

Whatever procedures are used to provide the shell design concept, design-development tests should be performed to verify the design and analysis.

Minimum gage requirements should be set by fabrication restraints or by the stipulation that the shell should not buckle under the design limit load or should not fail at ultimate load.

4.2.7.5 Joints

Static design of a bonded joint should begin with selection of the most efficient concept that fits within the design envelope and provides sufficient access for inspection. At present, the most efficient concept is the scarf joint or the step-lap joint, followed by the double-lap and single-lap joints in descending order. The designer should attempt to minimize eccentricity, avoid peel loads on the adhesive, and minimize changes in total stiffness across the joint. The design curves in references 14 and 82 should be used with caution beyond the range of actual test data. Reference 63 has shown significant differences in static strength between one-fifth-scale and half-scale replicas of a double-lap joint. None of the current analytical techniques fully accounts for such a scale effect, and full-scale static tests are necessary to ensure adequate strength.

The most efficient static design concept for a bolted joint appears to be the tapered multiple-fastener joint where the stiffness is varied to maintain equal distribution of loads among the fasteners. The procedure of reference 136 for estimating the distribution of bolt loads in such a joint is recommended.

The efficiency of a given joint configuration can usually be improved by forcing net tension, shear-out, and bearing failure modes to occur simultaneously. To achieve this goal for simple joints, the design curves in references 14 and 82 should be used to determine the required width-to-diameter ratio, edge-distance-to-diameter ratio, and thickness. Reference 133 has shown that the strength of a laminate with a circular hole varies with the hole diameter. When the design curves of references 14 and 82 are used for bolt diameters beyond the range tested, the scaling effect should be accounted for by analysis or test.

In general, the design practices that improve static joint efficiency also enhance the joint fatigue characteristics, and efforts should be directed toward minimizing changes in load path and in the number and severity of stress concentrations. Local reinforcing methods (e.g., metallic reinforcements, doublers, or local ply buildups) used to develop acceptable joint strengths also introduce additional local eccentricities which should be accounted for in the joint design.

Fail-safe design concepts should include multiple joints so that failure of any one will not degrade strength below a specified level. Because of the brittle failure characteristics of composites, caution should be exercised to ensure that load redistribution after the failure of one element can occur without overloading adjacent elements.

4.2.7.6 Component Design for Strength and Stiffness

Optimization procedures should be employed to achieve a high level of structural efficiency for both the deterministic and the reliability design approaches. For a reliability approach, the design procedure discussed in Section 4.2.4 should be used in sizing details to meet the overload, aeroelastic, and life-time requirements and the component reliability goal.

4.2.8 Producibility

The basic practices of reference 82 should be followed to ensure the producibility of boron-epoxy, graphite-epoxy, and boron-aluminum composites. Since this field is continually being updated and improved, the progress reports of current government and industry research programs should be consulted for new fabrication methods and techniques. Current production programs should also be used as sources of manufacturing producibility procedures.

4.2.9 Maintainability

The structural design should allow sufficient access to parts for inspection, repair, and replacement. Only judgment can be recommended as the means for balancing design simplicity and accessibility.

4.3 Fracture Control

4.3.1 Control Plan

One of the first tasks in a fracture-control plan should be to select the components that are critical to the completion of the mission. The second task should be to assign responsibilities for achieving fracture control to the organizations directly involved in the component design and fabrication. The third task should be to generate a data bank on behavior of laminates and structural elements with respect to static strength, fatigue life, and residual strength. The structural elements tested should be representative of the stress concentrations found in the selected components such as bonded joints, open holes, loaded holes, and noncircular cutouts. The fourth task should be to implement the quality assurance recommendations set forth in Section 4.4.4, particularly with respect to monitoring *in situ* variability of strength and fracture toughness. Reference 4 should be used to establish the overall content and intent of the fracture-control plan.

4.3.2 Design

4.3.2.1 Service-Life Philosophy

Each component should be evaluated to determine whether a safe-life or fail-safe approach is more appropriate in terms of mission requirements, cost, fabricability, and maintainability. Components that are especially vulnerable to damage are appropriate candidates for a fail-safe approach. The conventional approach of providing redundant elements should be considered as well as the integral buffer-strip concept presented in reference 140.

For a deterministic design approach, the definitions of safe-life and fail-safe currently being used for metallic structures (ref. 4) should be used.

For a reliability-based design approach, the following definitions should be used.

Safe-Life

A safe-life design should not allow a degradation in strength which would prevent the structure from meeting its reliability goals throughout the design service life.

Fail-Safe

For a fail-safe design, the residual strength after failure of a single structural element should permit operation at a specified reduced reliability until the next scheduled inspection when repairs can be made.

4.3.2.2 Stress Concentration Effects

For static design, the effect of large discontinuities such as access holes, windows, and cutouts on laminate strength should be determined with appropriate analytical procedures (finite-element and integral-equation computer procedures) in conjunction with a suitable failure theory. The effect of small discontinuities such as fastener holes on laminate strength should be evaluated empirically or predicted by a behavioral model which accounts for the variation in strength with discontinuity size (refs. 133 and 139). In general, the stress concentration factor of a hole increases with an increasing percentage of filaments in the load direction. For example, a longeron containing predominantly unidirectional fibers should not be penetrated with fastener holes. In fact, it is advisable to provide strain-tolerant areas around cutouts, attachments holes, and other stress concentrations in the laminate (ref. 119).

The degradation in strength and life due to random defects resulting from fabrication processes such as voids should be evaluated by test. The results should be used to establish quality-control standards.

Good design practices should minimize manufacturing defects. Design practices that cause high stress concentrations should be avoided. For example, bucked rivets should not be used for joining because the stress concentration caused by the hole is increased by the additional stresses from preloading the rivets. Complex designs that require many sequential fabrication operations should be avoided. Whenever feasible, composite assemblies should be laminated and cured in one step. For good composite design, a management-level effort should be made to achieve the simplest possible composite structure to meet functional objectives.

4.4 Design Verification

4.4.1 Material Procurement and Acceptance

4.4.1.1 Procurement

Material procurement specifications should be approved by the contracting agency before any composite materials are purchased. An example of the physical and mechanical properties that should be required is given for boron-epoxy in Section 2.4.1.1. Similar specifications should be written for all the composite materials to be used. In general, these specifications should include minimum values for the physical properties along with the test procedures by which the properties may be obtained. For the desired mechanical properties, statistical values should be specified, including an indicator of scatter for each, and the tests whereby those properties may be obtained should also be specified.

The prospective material suppliers should be required to conduct this series of tests on the candidate material before it is shipped to the buyer.

4.4.1.2 Acceptance

Material acceptance specifications should be approved by the contracting agency before any composite materials are bought. An example of the physical and mechanical properties that should be checked for boron-epoxy is given in Section 2.4.1.2 The physical property checks required at this point should be limited to dimensional checks on the material. As an example of these dimensional checks, the width of the material and the position of the material on the paper backing should be checked at various points for material bought in the form of 0.0762-m (3-in.) tape. The required mechanical properties should include the following lamina data obtained at room temperature and at one other appropriate temperature: $[0]_c$ and $[90]_c$ flexure strengths and horizontal shear strength. These properties should have a statistical basis, including a measure of scatter. The variation of mechanical properties should be monitored throughout the production cycle. The tests for these properties should be included in the acceptance specification for each material.

4.4.2 Structural Testing

4.4.2.1 Strength and Deformation

Structural qualification strength and deformation tests should be performed on the full-scale structure. The test fixture should realistically simulate the loads and environments that the structure will experience in flight. Sufficient measurements (e.g., temperature, strains, and deflections) should be recorded to verify that the structure is experiencing the proper input loads. These measurements should then be recorded at 20-percent load increments up to limit load in each loading condition. After the limit-load demonstrations, the structure should be loaded in 10-percent increments to ultimate load and then to failure in the critical load condition.

References 6, 8, and 146 provide information and procedures for static qualification tests (strength and deformation) on metallic structures. These procedures are applicable to composite material structures.

4.4.2.2 Life

For a deterministic design approach, the structure should exhibit a test life equal to the design fatigue life multiplied by the appropriate scatter factor. This testing should be performed with a random fatigue spectrum.

For a reliability design approach, two life verification tests should be conducted to satisfy the life reliability goals for the static overload and fatigue conditions. Appendix B recommends a method for conducting these tests.

4.4.2.3 Damage Tolerance

Damage-tolerance levels should be demonstrated through element or subcomponent tests. These tests should accurately represent both the loading situation and the damage to the structure. The tests may be either static or fatigue, depending on the anticipated critical failure modes. For fatigue tests, a random load spectrum should be used. For a deterministic design, the damaged structure should only be required to sustain a specified reduced load level. For a reliability-based design, the damaged structure should only be required to perform at a specified reduced reliability for the maximum inspection interval.

4.4.2.4 Dynamics

The ability of composite structures to withstand all anticipated dynamic loading conditions should be demonstrated with tests similar to those for metallic structures. References 7 and 149 give recommended practices.

4.4.3 Quality Assurance

4.4.3.1 Destructive Evaluation

Destructive testing should be performed on flight-critical parts. The first part cured on a new tool should be destructively tested, and subsequent parts should be similarly tested on a random basis. Destructive testing specifications should be approved by the contracting agency. These specifications should give the sampling technique for choosing the test parts and the testing criteria. Each of the selected parts should be inspected by nondestructive techniques (e.g., ultrasonics or radiography) before it is tested.

Another type of destructive evaluation as a means of quality assurance involves the testing of test tabs. Test tabs should be fabricated and tested to ensure the integrity of flight-critical parts. The test tabs should be integral with the part where feasible and should represent the part with respect to cure cycle, lay-up, material, and tooling. Standard flexure and horizontal shear specimens should be cut from these test tabs and tested to failure. The properties obtained from these specimens should compare with acceptable values.

4.4.3.2 In-Process Controls

Material properties should be maintained by in-process controls from acceptance through the cure cycle of the part. Since epoxy-preimpregnated tape is only partially cured, the material should be stored at low temperature in a sealed bag to retard further curing. A record should be kept of the total time at room temperature. The material should be allowed to stabilize at room temperature before it is removed from the sealed bag. After the material has been at room temperature for a specified time, it should satisfy the acceptance tests again before being used. In-process controls should also be exercised during the cure cycle. However, cure-cycle specifications are beyond the scope of this document.

The quality assurance practices in reference 82 should be followed for all composite material systems.

4.4.3.3 Acceptance Testing

Proof- or acceptance-testing should be designed to accomplish the following:

- Detection of errors in material processing and fabrication
- Detection of low-strength parts

The acceptance inspection of a composite part should include at least the following:

- Visual check of the part for gaps and wrinkles in the material

- Dimensional checks for compliance with drawings
- Thickness survey
- Hardness measurements taken along outer edges
- Evaluation of test tabs for flight-critical structure
- Nondestructive inspection (e.g., ultrasonics, radiography, etc.) of all flight hardware

Proof testing should be used to verify the structural design.

The ability of the acceptance test to determine material defects or fabrication errors is perhaps its most important characteristic. This error-disclosure capability is important for both fatigue-critical and overload-critical structure. The proof-test load level should be determined through knowledge of the structure gained from analysis or tests or from the material damage load levels.

4.4.3.4 In-Service Inspection

In-service inspections should be performed periodically throughout the life of the composite structure. These inspections should begin with a visual check for damage such as resin crazing or delaminations. Nondestructive inspection methods should be used to delineate the area of voids or delaminations. If accidental damage has produced holes in the laminate, the material adjacent to the holes should be checked for delaminations and resin crazing.

4.4.3.5 Repair

The patching of holes or damaged areas and the repair of delaminations are described in detail in references 151 and 152. The procedures given in these references should be followed for these two types of damage.

Appendix A
RELIABILITY-BASED DESIGN PROCEDURE

A generalized flow diagram for the reliability-based design procedure is shown in figure 22. This design procedure is based on requirements established by NASA or the contractor on the basis of (1) the planned use of the vehicle, and (2) experience with similar vehicles. These requirements include structural configuration, exceedance curves, overload conditions, the required life of the structure, the reliability goal to be satisfied at the end of the life, and the estimated size of the fleet. The reliability goal may be specified for the individual structure or the entire fleet, since they are related by the equation

$$P_{fleet} = P^n_{structure}$$

where P_{fleet} is the probability of survival of the fleet, $P_{structure}$ is the probability of survival for the individual structure in the fleet, and n is the number of structures in the fleet.

As specified in reference 68, a failure mode, effect, and criticality analysis should be performed on the preliminary configuration to identify structural components or locations that are critical to the safety of the structure. The number of critical points will depend on the structural complexity (e.g., a few for a simple structure and as many as 50 for a large, complex vehicle).

The reliability goal for each of these critical components, assuming that each critical failure location in a structure is completely independent, is

$$P_{component} = P_{structure}^{\left(\frac{1}{n_{component}}\right)}$$

where $P_{component}$ is the probability of survival of each critical component, $n_{component}$ is the number of critical components, and $P_{structure}$ is as defined previously. Estimates of the reliability of critical components should be based on test data.

The fail-safe elements of the structure are not critical, and must only be designed for the desired maintenance periods. For flight-critical parts, redundancy must be demonstrated. The reliability goal of the critical components should be apportioned to the design details that make up a component, such as laminates, laminates with holes, bolted joints, and bonded joints. This apportionment should take into account whether the failure of a design detail depends on or is independent of the failure of the other design details in a critical component.

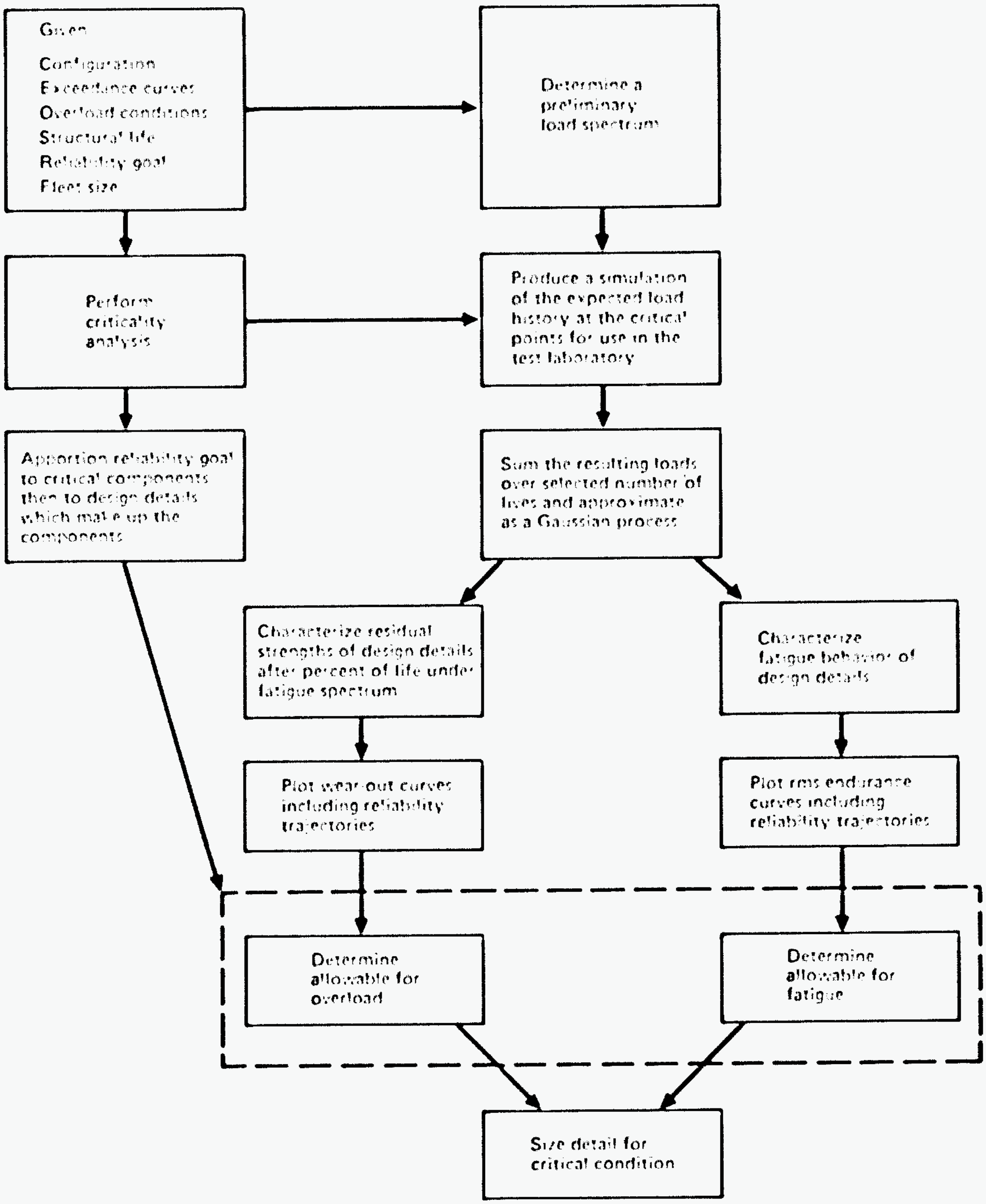


Figure 22. – Reliability-based design procedure.

At one extreme, the assumption of complete dependency, inferring redundancy or parallelism, results in a less severe reliability goal for each design detail. At the other extreme, the assumption of complete independence (details in series with one another), which will be mandatory with respect to maintenance-related reliability, results in a far more severe reliability goal for each design detail. In fact, this latter case emphasizes the penalty paid for complexity in a component, since the reliability requirement for a design detail increases as the number of independent details increases,

$$P_{detail} = \left[P_{component}\right]^{\left(\frac{1}{n_{details}}\right)}$$

where P_{detail} is the probability of survival of the design detail and n_{detail} is the number of design details in the particular component. It is recommended that the reliability goal for each design detail (for which the level of compliance will be experimentally determined) be set equal to the reliability goal for the critical component. Maintenance certification should be made on the basis of independence. The results of this apportionment procedure will be used later in the design process.

The basis of the recommended design process is the experimental characterization of design concepts with respect to the expected service life of the vehicle. Thus, definition of the expected service life is the next step in the process. A preliminary load spectrum should be determined on a mission segment basis. For initial design purposes, this spectrum may be based on the documented usage of similar vehicles. The spectrum is used to produce a life-cycle digital tape or a laboratory computer simulation of the expected variations in loads with time in the critical areas of the structure. A procedure for this step can be found in reference 63. The high-load tail of the resulting curve of load vs. time can be approximated by a Gaussian process. The Gaussian process is completely defined by its mean load, μ_P (normally the 1-g condition), and its root-mean-square load, σ_P.

As shown in figure 22, the load spectrum is used to characterize experimentally the candidate design details in two similar ways, one path resulting in the assessment of the overload capability and the other providing fatigue-life information. A common requirement for the two paths is the application of test loads to specimens of the design details. This is done by subjecting sets of specimens to sets of loads which are linear multiples of the given spectrum; for instance,

$$S_i(t) = C_i\, S(t) = C_i \frac{P(t)}{A}$$

where $S_i(t)$ is the test specimen stress, C_i is a set of constants ranging from, say, 0.7 to 1.3, $S(t)$ is the reference stress, $P(t)$ is the load prescribed by the spectrum, and A is the test specimen area. The subscript i is the number of sets of loads or spectrum perturbations, and t is time. The sets of stress variations of the test specimen with time may each be characterized by their mean stresses

$$\mu_{si} = C_i \mu_p / A$$

and their root-mean-square (rms) stresses

$$\sigma_{si} = C_i \sigma_p / A$$

For fatigue-life characterization, sets of specimens are cycled to failure at various rms stress levels. These results are plotted as shown in figure 23. This plot then provides the distribution of fatigue life for a range of rms stress levels. Statistical procedures are

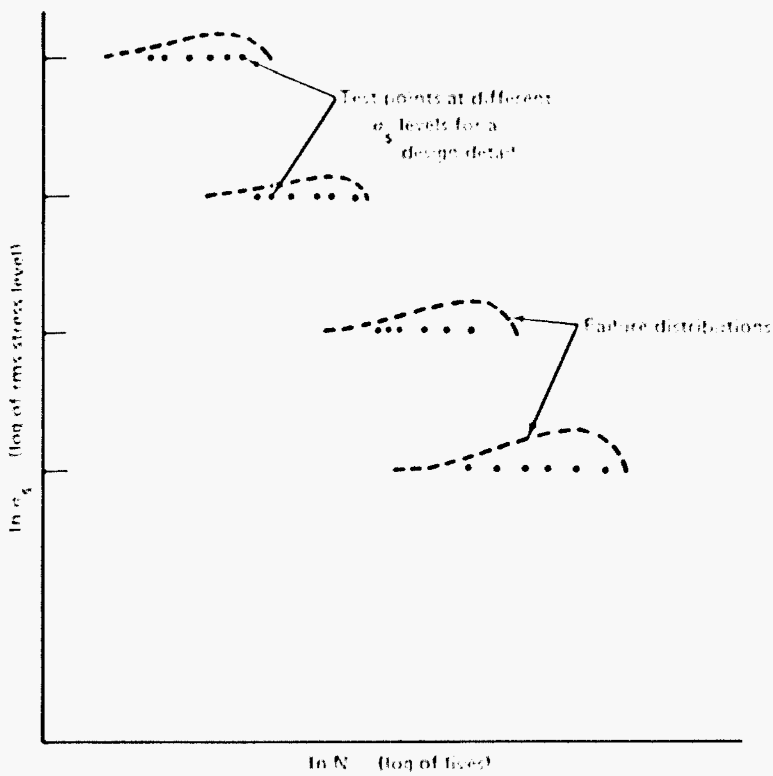


Figure 23. – Root-mean-square stress versus endurance

then used to define probability-of-survival trajectories for fatigue behavior, as shown in figure 24. The allowable rms stress for fatigue is subsequently obtained by entering figure 24 with the required structural life and the previously calculated reliability goal for the design detail.

The overload characterization for the design details uses the experimental equipment in a slightly different manner. For a particular rms stress level, a set of specimens is subjected to a percentage of the spectrum lifetime. These specimens are then static tested to determine their residual strengths. Repetition of this procedure at various percentages of lifetime produces the data required to plot a residual strength or wear-out curve. Again through statistical procedures, reliability trajectories may be superimposed on the wear-out curves shown in figure 25, where typical data are presented as an average value plus a range of values. A set of wear-out curves corresponding to a range of rms stress levels must be obtained. Figure 26 shows the effect of a lower rms stress level, which is denoted by the the dashed reliability trajectories. This comparison indicates that a small decrease in rms stress provides a tremendous reliability improvement. The next step in the procedure is to relate the information contained in the wear-out curve to the required reliability goal.

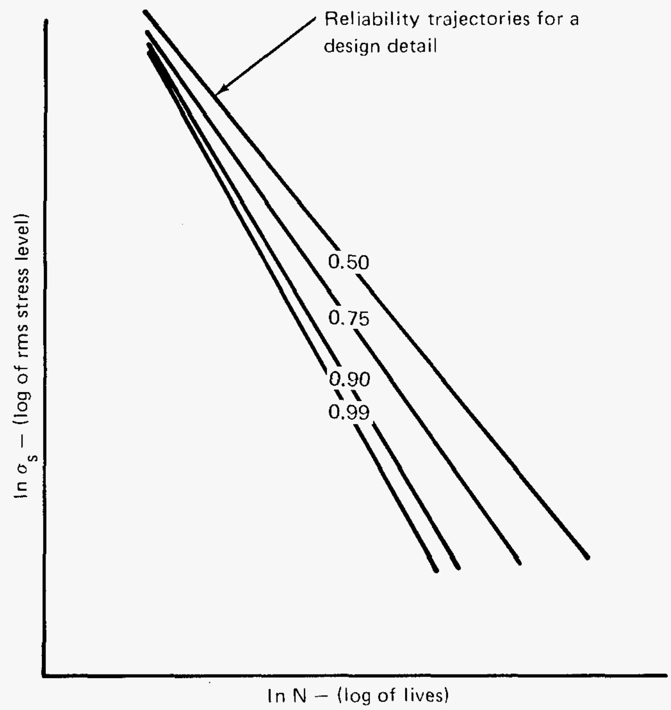


Figure 24. – Root-mean-square stress versus endurance with reliability trajectories.

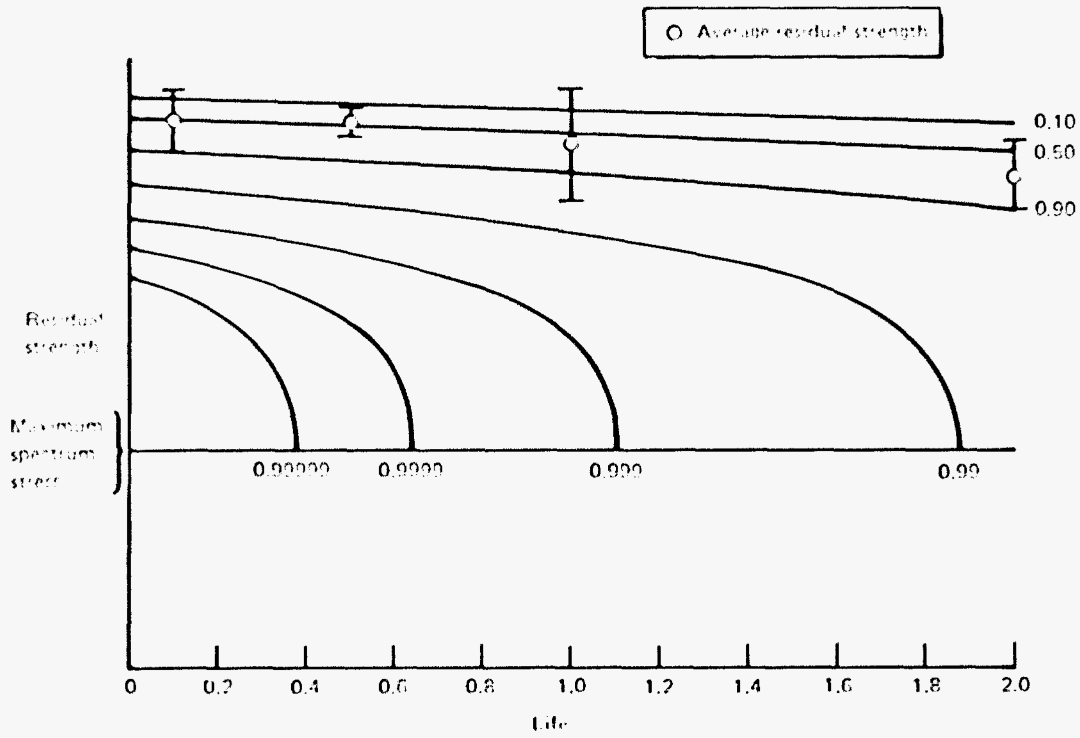


Figure 25. – Wear-out curve with reliability trajectories.

With the wear-out curves of figure 26, the probability of surviving the overload condition can be determined as a function of time or life. Figure 27 shows two such plots for the two rms stress levels and the overload shown in figure 26. Since an overload reliability goal is to be satisfied for some specified lifetime, the reliability gained throughout life by the reduced rms stress level is apparent in figure 27. Various rms stress levels may be tried until the sought-after reliability is attained at the specified life. When this rms stress level is related to an area of thickness of the part, the result is a design which will have a specified reliability at the end of a prescribed lifetime. In addition, specified *stiffness requirements* for unusual conditions such as aeroelastic or thermoelastic effects should be accounted for at this point.

The design detail is now sized using the more critical of the two conditions (fatigue or overload), and thus will meet its reliability goal in the more critical of the two failure modes, fatigue or overload, and will surpass its reliability goal in the other.

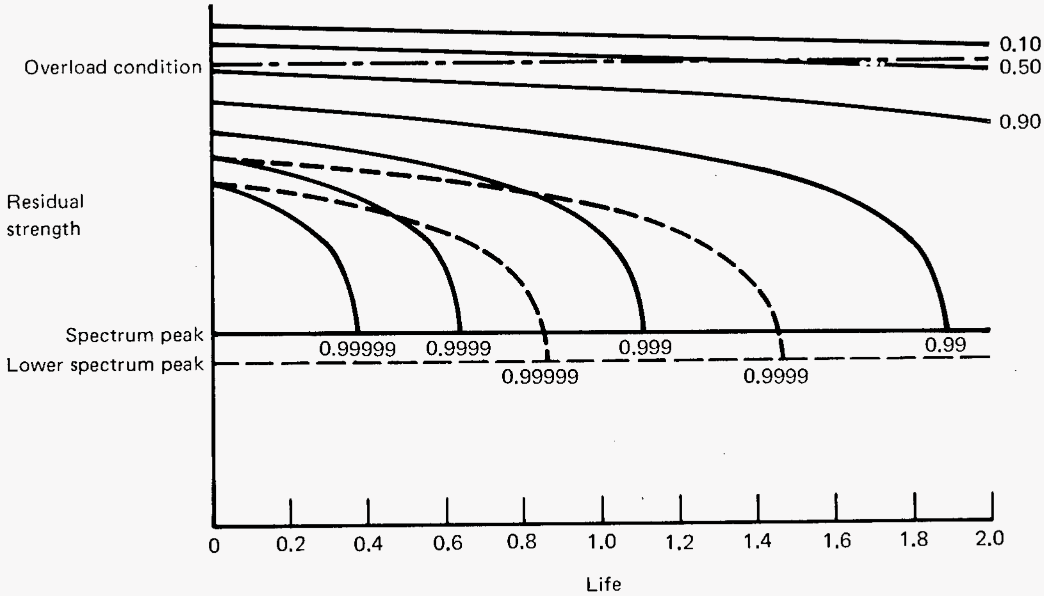


Figure 26. – Effect of stress level on reliability.

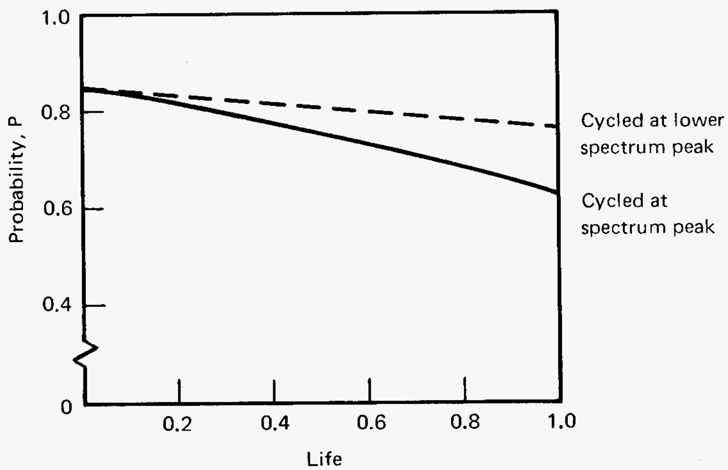


Figure 27. – Probability of survival versus life under two spectra.

Appendix B
LIFE VERIFICATION TESTING FOR RELIABILITY-BASED DESIGN

The two life requirements for a reliability design approach may be written as follows:

$$R_S \geqslant SRG$$

and

$$R_F \geqslant SRG$$

where

R_S = static overload reliability goal at the end of one service life

R_F = fatigue reliability goal at the end of one service life

SRG = structural reliability goal for total vehicle

Figure 28 shows the relationship between the various reliability goals at the end of one lifetime. The shape of the distributions shown in figure 28 should be the same as the corresponding subcomponent or element tests for similar failure modes. In order to establish the position of these curves, two types of tests should be considered: (1) fatigue test to failure under the random load spectra (see Sec. 4.2.1), and (2) fatigue test to one life and then static test to failure. A minimum of one each of these tests should be conducted. These single test results should then be considered as one-point estimates of the means of the distribution for the component. The primary structural failure mode and its anticipated distribution should be analyzed. After the appropriate penalty is paid for the restricted sample size, the expected reliability of the structure should be compared to the reliability goal. Two routes are available in case of deficiency. First, the structure may be modified and a recertification attempted; or second, a better estimate of the component mean capacity or scatter may be obtained.

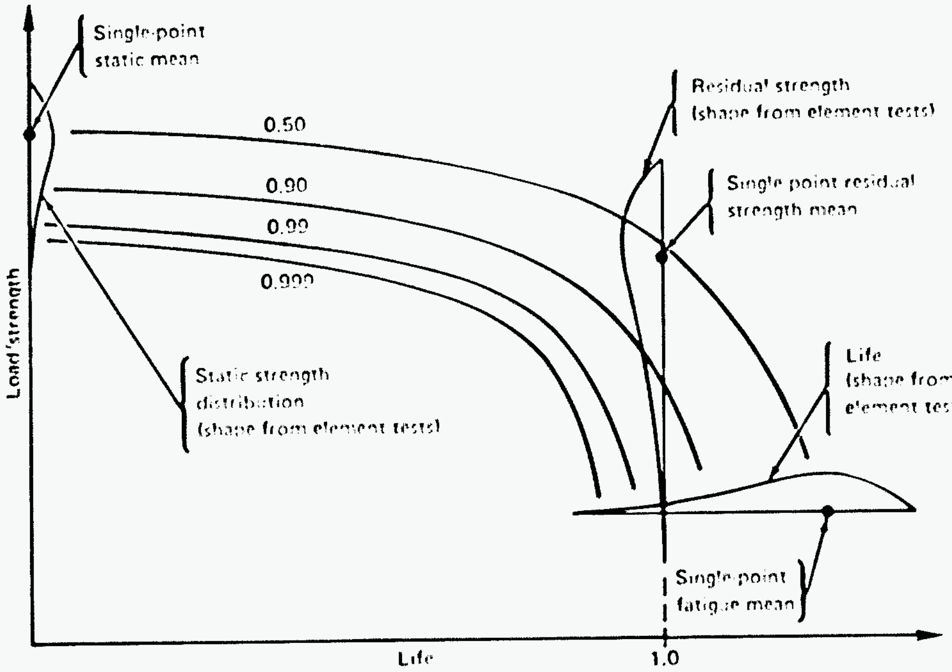


Figure 28. – Component residual strength and life distributions.

Appendix C
DESIGN, TOOLING, AND FABRICATION DETAILS

The following information on design, tooling, and fabrication may be used to achieve a better, more uniform part constructed with composite materials. These are design hints to and from the designer on the board. Some of these items have been included in Section 3.

C.1 Design

C.1.1 Unbalanced Laminates

Thin laminates will curl or twist severely if unbalanced and, even though they may be forced back with little pressure, the problems involved in holding them during trim, prefit, drilling, adhesive lay-up, or bond cure will usually result in a discrepant part at some stage of manufacture.

Thick laminates will not warp as severely as thin laminates, but the forces required to pull them back to shape are proportionately greater, and may result in warping the mating structure rather than pulling the warped part back to shape.

Unbalanced areas of a local nature such as in the taper of laminate thickness are usually acceptable, but should be minimized as much as possible.

C.1.2 Mixtures of Materials and Resin Systems

Boron-graphite mixtures should be specified with caution because the resin bleed rates are different and may cause one material to be resin starved while the other is resin-rich, resulting in poor laminate properties.

Mixtures of materials with different resin systems should also be treated cautiously because poor interlaminar strengths may cause the laminate to split during cool-down or handling.

C.1.3 Drawings of Composite Parts

The laminate code for sandwich panels should be defined as the way one would see the part on the tool. Only the 0-deg direction should be shown on the drawing. The plus and minus directions should be shown only to denote a change in the convention that positive angles are measured clockwise from the 0-deg direction. (If a reversal in the positive angle direction is needed for an opposite-hand part, it should be shown in a sketch in the general notes of the drawing.)

C.1.4 Edge and Cap Material

Parts with many edge buildups and tapers in thickness should be drawn on multiple sheets or with multiple views to define the perimeter of each ply adequately. Ozalid copies or photo reproductions of the shape of the basic part will aid in this, and the time expended will be more than compensated for in a lower material scrap rate and lower lay-up time.

Cap material for a frame or bulkhead should be inserted in the basic skin where possible.

C.1.5 Laminate Coding

The laminate coding should be preceded by the total number of plies in the laminate. If the laminate tapers, the maximum thickness should be coded and then the plies to be dropped should be defined on the drawing. (The ply count is a simple shop check to determine if the laminate code has been translated properly.)

On drawings where separate laminates are laid up as details and placed into a larger laminate, a sequence chart should be prepared and placed on the first drawing sheet near the general notes showing the lay-up sequence.

C.2 Tooling

C.2.1 Tool Design

Tools should be designed to be compatible with the coefficient of expansion of the part to be built. They should be lightweight tools capable of fast heating rates. When stability is required, a tool should be of sandwich construction with either graphite skins for graphite parts or glass skins for boron parts. Other tools are adequate for parts that do not have to have a coefficient of expansion compatible with that of the composite. A solid-glass fabric tool is adequate for production runs of both graphite and boron parts when thermal compatibility is not required. The less expensive plastic-faced plaster tool used for boron and graphite parts is useful only for limited production runs.

C.2.2 Tools for Graphite Frames or Flanged Webs

A laminate may be cured on a male tool with only a blanket or autoclave bag to provide pressure; however, a female tool requires a rather expensive cast-rubber punch to compact the laminate in the radius area where the flange turns into the web of the part (and still may require local repair). The minimum radius for a female tool should be at least 5.588 mm (0.22 in.), and a larger radius is to be preferred.

C.3 Fabrication

C.3.1 Bonding Honeycomb Core

The tapered areas from the edge member to the center thickness of a honeycomb panel are irregular and difficult to prefit to the core (especially if the skin has been laid up by hand). Also, the bag surface is irregular, and a double layer of adhesive is usually required to produce a satisfactory bond.

C.3.2 Trapping a Laminate Between Two Parts

The irregular surface of a graphite laminate will require a greater thickness tolerance allowance if the part is to be trapped between two other parts than the average per-ply thickness indicates. The irregularities may be due to bag wrinkles, wrinkles in material due to compaction onto a male tool, wrinkles where continuous filaments cross a buildup, etc. These surface irregularities are usually local and normally will not affect fastener grip lengths, but they can prevent two parts from seating properly if not provided for.

C.3.3 Dimensions

The method of manufacture must be considered when the ply trim of a laminate is dimensioned. A tolerance of ±1.524 mm (±0.06 in.) is about the best that can be reasonably held in machine lay-up; however, it is questionable for hand lay-up on contoured parts.

C.3.4 Buildups

A method should be developed to maintain the configuration during lay-up and cure of built-up parts such as longerons or bulkhead caps.

C.3.5 Glass Tie-Plies

When attachments are added to an already cured skin (e.g., the insertion of a cured frame), a wet rub-out glass-tie-ply should be used where possible rather than a tie-ply which requires high pressure.

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SP-8005	(Environment)	Solar Electromagnetic Radiation, June 1965–Revised May 1971
SP-8006	(Structures)	Local Steady Aerodynamic Loads During Launch and Exit, May 1965
SP-8007	(Structures)	Buckling of Thin-Walled Circular Cylinders, September 1965–Revised August 1968
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