COMPOSITE AIRFRAME STRUCTURES

THIRD EDITION

PRACTICAL DESIGN INFORMATION AND DATA

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GLOSSARY

A

ADHEREND: The material being bonded by an adhesive.

ADHESION: The property denoting the ability of a material to resist delamination or separation into two or more layers.

ADHESIVE: A substance capable of holding two materials together by surface attachment. In the composites, the term is used specifically to designate structural adhesives, those which produce attachments capable of transmitting significant structural loads.

ADHESIVE FAILURE: Failure at the adhesive/adherend interface.

AGING: The effect on materials of exposure to an air environment for an interval of time.

AMORPHOUS: A material, such as a liquid, which has no crystallinity; i.e., there is no order or pattern to the distribution of the molecules in the material.

ANELASTICITY: A characteristic exhibited by certain materials in which strain is a function of both stress and time, such that while no permanent deformations are involved, a finite time is required to establish equilibrium between stress and strain in both the loading and unloading directions.

ANISOTROPIC: Not isotropic; having mechanical and/or physical properties which vary with direction relative to natural reference axes inherent in the material.

ANNEALING: In plastic, heating to a temperature where the molecules have significant mobility, permitting them to reorient to a configuration having less residual stress.

ARAMID: A type of highly oriented organic material derived from polyamide (nylon) but incorporating aromatic ring structure.

AREAL WEIGHT: The weight of fiber per unit area (width × length) of tape or fabric.

ARTIFICIAL WEATHERING: Exposure to laboratory conditions which may be cyclic, involving changes in temperature, relative humidity, radiant energy, and any other elements found in the atmosphere in various geographical areas.

AUTOCLAVE: A closed vessel which applies pressure to objects inside, such as a bagged laminate. The pressurizing medium is a gas (usually nitrogen or carbon dioxide). "Hydroclave" is an autoclave, except that water is used as the pressurizing medium.

AUTOCLAVE MOLDING: A process in which the layup is covered by a pressure bag, and the entire assembly is placed in an autoclave capable of providing heat and pressure for curing the part. The pressure bag is normally vented to the outside.

B

B-STAGE: An intermediate stage in the reaction of a resin; that is, partial cure.

BAGGING: The process of applying an impermeable layer of film over a part and sealing the edges so that a vacuum can be drawn. The bag permits a pressure differential to exist between the pressurizing medium (usually the working fluid of the autoclave or hydroclave) and the part, thereby applying pressure to the part.

BALANCED LAMINATE: A composite laminate in which all laminate at angles other than 0° and 90° occur only in pairs (not necessarily adjacent), and is symmetrical about a centerline. See Symmetrical Laminate.

BARRIER FILM: The layer of film used to permit removal of air and volatiles from a composite layup during cure while minimizing resin loss.

BATCH (or LOT): In general, a quantity of material formed during the same process and having identical characteristics throughout. As applied to the composites, a batch of prepreg tape is defined as a quantity of tape which is produced from a single batch of matrix material. The prepreg tape batch is not necessarily produced at one time, but all sub-batches are produced in the same equipment under identical conditions. The filaments included in a batch, in this context, are restricted only to being acceptable within the requirements of applicable filament procurement specifications.

BINDER: A bonding resin used to hold strands together in a mat or preform during manufacture of a molded object.

BISMALEIMIDES (BMI): Lower temperature capability addition polyimide resin systems which have epoxy-like processability but have improved elevated temperature properties.

BLEEDER CLOTH: Material, such as fiberglass, used in the manufacture of composite parts to allow the escape of excess gas and resin during cure. The bleeder cloth is removed after the curing process and is not part of the final composite.

BOND: The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

BRAIDING: Weaving of fibers into a tubular shape instead of flat fabric. This technique is currently used in the fabrication of graphite fiber reinforced golf shafts. It is being considered for aerospace tubular structures.

BREATHER CLOTH: A layer or layers of open weave cloth used to enable the vacuum to reach the area over the laminate being cured, such that volatiles and air can be removed and also causing the pressure differential that results in application of pressure to the part being cured.

BRIDGING: Special techniques must be used so that the fibers will move into radii and corners; otherwise, they "bridge" the gap, resulting in dimensional control problems and voids. Care must also be taken to prevent bridging of separators, bleeders, barrier films, venting layers, and bagging.

BRITTLE: Most of the high performance resin systems are brittle; namely, they exhibit very low elongation and almost no plastic deformation prior to failure. In composites, the mismatch in thermal coefficients of expansion between the fibers and the resin matrix, in combination with a brittle resin, often results in formation of microcracks.

BROADGOODS: A term loosely applied to prepreg material greater than about 12 inches in width, usually furnished by suppliers in continuous rolls. The term is currently used to designate both collimated uniaxial tape and woven fabric prepregs.

BUCKLING (COMPOSITE): Buckling is a mode of failure characterized generally by an unstable lateral deflection caused by compressive or shear action on the structural element involved. In composites, buckling may take the form not only of conventional general instability and local instability, but also a microinstability of individual fibers.

C

C-SCAN: The back and forth scanning of a specimen with ultrasonics. An NDT (nondestructive test) technique for finding voids, delaminations, defects in fiber distribution, etc.

CARBON: The element which provides the backbone for all organic polymers. Graphite is a more ordered form of carbon; diamond is the densest crystalline form of carbon. Most of the high strength "graphite" fibers are actually carbon fibers; with higher temperature processing, the same organic precursor fiber can be converted to the higher modulus graphitized form.

CARBON FIBERS: Fibers made from a precursor by oxidation and carbonization, and not having a graphitic structure.

CATALYST: A chemical which promotes a chemical reaction without becoming a part of the molecular structure of the product. In resin systems, catalysts (and accelerators) lower the temperature at which significant amounts of reaction occur, affecting reaction rate and changing the characteristics of the cure cycle.

CAUL PLATES: Smooth plates, free of surface defects, used during the curing process to provide a controlled surface on the finished laminate.

CELANESE COMPRESSION: A specialized compression test using a fixture designed by Celanese Corp. Now a standard ASTM test.

CERAMIC TOOLING: Use of a castable ceramic to make a tool shape. Ceramic tooling is seldom used unless a very large number of complex parts are to be made; otherwise, tooling such as graphite tooling is more cost-effective.

CHROMATOGRAM: A plot of detector response against peak volume of solution (eluate) emerging from the system for each of the constituents which have been separated.

CLOTH: A woven product made from continuous yarns or tows of fiber. "Cloth" and "fabric" are usually used interchangeably.

COCURING: The act of curing a composite laminate and simultaneously bonding it to some other prepared surface during the same cure cycle.

COEFFICIENT OF THERMAL EXPANSION: The change in a dimension of a specimen per unit change in temperature, expressed as a ratio. Typical units are microinches/inch/°F and microinches/inch/°C. High modulus graphite fibers have a negative coefficient of thermal expansion (CTE) in the axial direction; they shrink when heated. The CTE only describes the reversible portion of the dimensional change.

COEFFICIENT OF VARIATION: The standard deviation divided by the mean.

COHESIVE FAILURE: Describing the failure surface of an adhesive joint where the failure occurred primarily in the adhesive layer.

COMPLEX CURVATURE: Describing a surface which curves in more than one direction, such as a saddle or spherical shape. In other words, the surface has both concave and convex areas.

COMPOSITE: As used generally in this book, composite describes a matrix material reinforced with continuous filaments.

COMPOUND: An intimate mixture of polymer or polymers with all the materials necessary for the finished product.

COMPRESSION MOLDING: Putting a reinforced resin into a mold cavity, closing the mold, and applying pressure and heat in order to force the material to completely fill the mold cavity and to cure the material.

CONDITIONING: Maintaining the test specimens in a controlled environment for a specific length of time prior to testing.

CONSOLIDATION: In metal matrix or thermplastic composites, the diffusion bonding operation in which an oriented stack of preplies is transformed into a finished composite laminate.

CONTINUOUS FILAMENT YARN: Yarn formed by combining two or more continuous filaments into a single, continuous strand.

CRAZING: The development of a multitude of very fine cracks in the matrix material.

CREEL: The rack used to hold fiber spools so that a large number of fiber tows can be used simultaneously, as in making tape prepreg or for the warp fiber tows of a weaving loom.

CROSSLINKING: Chemical reaction between molecules resulting in the formation of a three-dimensional network of molecules. Crosslinking requires that at least one of the molecules involved in the reaction have three or more reactive groups; otherwise, the reaction only results in forming a longer molecule (chain extension).

CROSS-PLIED: A laminate with plies in different directions. Laminates made from fabric automatically have fibers in two directions, but are not considered as "cross-plied" unless the different layers (plies) are oriented in different directions.

CRYSTALLINITY: Polymers such as nylon form localized area of crystallinity (highly ordered sections) formed by alignment of sections of a polymer chain (by folding, etc.) or of adjacent molecules. The localized areas of crystallinity change the physical behavior of the polymer.

CURE: To change the properties of a thermosetting resin irreversibly by chemical reaction. Cure may be accomplished by addition of curing agents, with or without catalyst, and with or without heat and pressure.

CURE CYCLE: The time/temperature/pressure cycle used to cure a composite resin system or prepreg.

CURE MONITORING: Use of electrical techniques to detect changes in the electrical properties and/or mobility of the resin molecules during cure.

CURE STRESS: A residual internal stress produced during the cure cycle of composites containing reinforcements and/or resins with different thermal coefficients of expansion.

D

DAM: Boundary support used to prevent excessive edge bleeding of a laminate and to prevent crowning of the bag.

DEBOND: A deliberate separation of a bonded joint or interface, usually for repair or rework purposes.

DEBULKING: Compacting the thickness of a layer of prepreg or of a prepreg layup by using pressure and/or vaccum to remove most of the air.

DELAMINATION: The separation of one or more layers of a laminate.

DENIER: A textile term for the weight, in grams, of 9000 meters of fiber tow.

DESORPTION: A process in which an absorbed material is released from another material. Desorption is the reverse of absorption, adsorption, or both.

DIELECTROMETRY: Use of electrical techniques to measure the changes in loss factor (dissipation) and in capacitance during cure of the resin in a laminate.

DIFFERENTIAL SCANNING CALORIMETRY (DSC): Measurement of the energy absorbed (endotherm) or produced (exotherm) as a resin system is cured. Also detects loss of solvents and other volatiles.

DIFFERENTIAL THERMAL ANALYSIS (DTA): An experimental analysis technique where a specimen and a control are heated simultaneously and the difference in their temperatures is monitored. The difference in temperature provides information on relative heat capacities, presence of solvents, changes in structure (i.e., phase changes such as melting of one component in a resin system), and chemical reactions. Also see Differential Scanning Calorimetry.

DISBOND: A lack of proper adhesion in a bonded joint. This may be local or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

DISSIPATION FACTOR: A measure of the lag in phase angle caused by presence of a material between the plates of a condensor (capacitor). Also called power factor, loss tangent, and tangent δ. In dynamic dielectric analysis of a resin undergoing cure, it is a measure of the energy used in aligning (or attempting to align) the dipoles present in the resin system with the constantly reversing (AC) electrical field. When the resin voscosity is low, it takes very little energy to align the dipoles; the dissipation factor increases as resin viscosity increases, but decreases when the resin is cured (when cured, the dipoles are immobilized, and very little power is lost).

DISTORTION: In fabric, the displacement of fill fiber from the 90° angle (right angle) relative to the warp fiber. In a laminate, the displacement of the fibers (especially in radii), relative to their idealized location, due to motion during layup and cure.

DRAPE: The ability of a prepreg to conform to a contoured surface. If the resin becomes hard, due to loss of solvent or due to staging, the prepreg becomes stiff (''boardy'') and loses its drape characteristics.

DRYING TOWER: In prepregging via a solvent process, a conveyor belt carries the prepreg through a drying section which uses heated air to remove excess solvent from the prepreg. Usually this heated section is vertical, due to space limitations.

E

EDGE BLEED: Removal of volatiles and excess resin through the edge of the laminate, as in matched die molding of a laminate. In autoclaved parts, edge bleeding is discouraged, since excess resin will only be removed from the area near an edge; resulting in uneven resin distribution.

ELASTOMERIC TOOLING: A tooling system utilizing the thermal expansion of rubber materials to form composite hardware during cure.

ENVELOPE BAG: A vacuum bag which encloses the laminate and the tool.

EXPANDABLE TOOLING: Use of hollow rubber mandrel which can be pressurized to form composite hardware during cure.

F

FABRIC: A material constructed of interlaced yarns, fibers, or filaments, usually arranged in a planar structure. Nonwovens are sometimes included in this classification.

FAYING SURFACE: The portion of a component's surface that, upon assembly, will be pressed against another component and hence must be pre-cleaned and otherwise treated for bonding.

FIBER: A single homogeneous strand of material, essentially one-dimensional in the macrobehavior sense, used as a principal constituent in composites because of its high axial strength and modulus.

FIBER CONTENT: The amount of fiber present in a composite. This is usually expressed as a percentage volume fraction or weight fraction of a cured composite.

FIBER DIRECTION: The orientation or alignment of the longitudinal axis of the fiber relative to a stated reference axis.

FIBER FINISH: A material applied to the surface of fibers to improve the bond of the fiber to the resin matrix and/or to protect the fiber against abrasion damage during operations such as weaving.

FIBER TOW: A loose, untwisted bundle of continuous fibers. In composite technology, "tow" is often used interchangeably with "yarn", the twisted version.

FIBER VOLUME: The volume percent of fiber in a composite.

FIBERGLASS: The generic name for glass fibers and for composites using glass fibers for reinforcement.

FILAMENT: Fibers characterized by extreme length, such that there are normally no filament ends within a part except at geometric discontinuities. Filaments are used in filamentary composites and are also used in filament winding processes which require long continuous strands.

FILAMENT WINDING: An automated process in which continuous filament (or tape) is treated with resin and wound on a removable mandrel in a prescribed pattern.

FILAMENT WOUND: Pertaining to an object created by the filament winding method of fabrication.

FILL: Yarn oriented at right angles to the warp in a woven fabric.

FILLER: A second material added to a basic material to alter its physical, mechanical, thermal, or electrical properties. Sometimes used specifically to mean particulate additives.

FILLER PLY: Partial plies, usually located on sandwich edgebands, which do not extend onto any portion of the honeycomb surface.

FILM ADHESIVE: Adhesive supplied in the form of a film.

FINISH: A material, with which filaments are treated, which contains a coupling agent to improve the bond between the filament surface and the resin matrix in a composite material. In addition, finishes often contain ingredients which provide lubricity to the filament surface, preventing abrasive damage during handling, and a binder which promotes strand integrity and facilitates packing of the filaments.

FLAME-SPRAYED TAPE: A form of metal matrix preply in which the fiber system is held in place on a foil sheet of matrix alloy by a metallic flame-spray deposit. Each flame-sprayed preply is usually combined in the layup stack with a metal cover foil and/or additional metal powder to ensure complete encapsulation of the fibers. During consolidation, all the metallic constituents are coalesced into a homogeneous matrix.

FLASH: Excess resin which forms at the parting line of a mold or die, or which is extruded from a closed mold.

FLEXURAL STRENGTH: A mechanical test to measure the strength of a laminate in bending. The strength reported is a calculated value which assumes that the material is isotropic in the thickness direction, which is approximately true for unidirectional specimens but is definitely not true for cross-plied laminates. In addition, failure modes can be in compression, in tension, in interlaminar shear, or a combination thereof. Therefore, the test is only suitable as a comparison or quality control test.

G

GEL PERMEATION CHROMATOGRAPHY (GPC): A form of liquid chromatography in which the polymer molecules are separated by their ability or inability to penetrate the material in the separation column.

GEL TEMPERATURE: In a cure cycle, the temperature at which the viscosity of a thermosetting resin becomes so high that no further dimensional change occurs. The temperature at which the resin gels can be changed by changing the cure cycle (heatup rate, hold times, etc.). The laminate dimensions become fixed at the gel temperature (for all practical purposes); hence, the tool dimensions at the gel temperature control the dimensions of the cured laminate.

GEL TIME: The amount of time required before a resin sample advances to the gelation point, when held at a pre-defined constant temperature.

GELATION: In composite technology, referring to the point in a resin cure where the resin viscosity has increased to the point where it barely moves when probed with a sharp instrument.

GELCOAT: A resin applied to the mold to provide an improved surface for the composite.

GLASS: An inorganic product of fusion which has cooled to a rigid condition without crystallizing. In the composites, all reference to glass will be to tne fibrous form as used in filaments, woven fabric, yarns, mats, chopped fibers, etc.

GLASS CLOTH: Woven glass fiber material (see also SCRIM).

GLASS TRANSITION TEMPERATURE (T_g): One method of describing the temperature at which increased molecular mobility results in significant changes in the properties of a cured resin system. The glass transition temperature (T_g) can be defined as the inflection point on a plot of modulus vs. temperature. T_g is defined as the inflection point — properties can have decreased significantly before T_g is reached.

GRAPHITE: The crystalline, allotropic form of carbon. In bulk form, used for advanced composite tooling and for such items as the "lead" in pencils. See Graphite Fibers.

GRAPHITE FIBERS: Technically, a highly oriented form of graphite. However, in common usage it also includes highly oriented carbon fibers which have only a small amount of graphite content.

GRAPHITIZATION: Conversion of carbon to its crystalline allotropic form by use of very high temperatures. Diamond is also crystalline allotropic form of carbon, but requires extremely high pressures (over one million psi) in addition to very high temperature in order to be formed.

H

HAND LAYUP: A process in which components are applied either to the mold or on a working surface, and the successive plies are built up and worked by hand.

HARDENER: The compound which reacts with a resin to form the crosslinked (thermoset) plastic.

HARDNESS: Resistance to deformation; usually measured by indentation. Types of standard tests include Brinell, Barcol, and Rockwell.

HARNESS SATIN: Describes a set of weaving patterns which produce a fabric having a satin appearance. "8HS" describes a harness satin weave where the warp fiber tows go over seven fill tows and then under one fill tow, for a repeating total of 8. By itself, "8HS" is not a complete description, because there are many possible patterns of where the crossover points of adjacent tows are located.

HOT MELT PROCESSING: Refers to the process of heating a resin to reduce its viscosity, using a doctor blade arrangement to spread a controlled thickness layer onto transfer paper, and subsequently forcing the hot resin (briefly heated to a higher temperature to reduce the viscosity) into collimated fibers. The process can also be used to prepreg fabric, but care has to be taken to avoid crushing the fibers where they cross over.

HYBRID: A composite laminate or prepreg contianing two or more types of composite systems. Usually only the fibers differ since cocuring two different resins is difficult.

I

INCLUSION: A physical and mechanical discontinuity occurring within a material or part, usually consisting of solid, encapsulated foreign material. Inclusions are often capable of transmitting some structural stresses and energy fields, but in a noticeably different degree from the parent material.

INTEGRAL COMPOSITE STRUCTURE: Composite structure in which several structural elements, which would conventionally be assembled together by bonding or mechanical fasteners after separate fabrication, are instead laid up and cured as a single, complex, continuous structure; e.g., spars, ribs, and one stiffened cover of a wing box fabricated as a single integral part. The term is sometimes applied more loosely to any composite structure not assembled by mechanical fasteners.

INTEGRALLY HEATED: Referring to tooling which is self-heating, through use of electrical heaters such as cal rods. Most hydroclave tooling is integrally heated; some autoclave tooling is integrally heated to compensate for thick sections, to provide higher heatup rates, or to permit processing at a higher temperature than the capability of the autoclave.

INTERFACE: The boundary between the individual, physically distinguishable constituents of a composite.

INTERLAMINAR SHEAR (ILS): Ideally, test methods used to measure ILS apply a pure shear load to the interface between two plies of a composite. The short beam shear (SBS) test does not apply a pure shear load; SBS strength values do not relate directly to ILS strength, but are suitable for quality control purposes.

INTERPLY: Two or more different reinforcements are combined in discrete layers and fibers are not mixed within a layer.

INTRAPLY: When reinforcements are mixed within a layer such as alternating strands in a fabric.

ISOTROPIC: Having uniform properties in all directions. The measured properties of an isotropic material are independent of the axis of testing.

K

KEVLAR: An organic polymer composed of aromatic polyamides having a pare-type orientation. (Parallel chain extending bonds from each aromatic nucleus.)

L

LAMINA: A single ply or layer in a laminate made of a series of layers.

LAMINAE: Plural of lamina.

LAMINATE: A produce made by bonding together two or more layers or laminae of material or materials.

LAMINATE ORIENTATION: The configuration of a crossplied composite laminate with regard to the angles of crossplying, the number of laminae at each angle, and the exact sequence of the lamina layup.

LAYUP: A process of fabrication involving the assembly of successive layers of resin impregnated material.

M

MACRO: In relation to composites, denotes the gross properties of a composite as a structural element but does not consider the individual properties or itentity of the constituents.

MANDREL: A form fixture or male mold used for the base in the production of a part by layup or filament winding.

MAT: A fibrous material consisting of randomly oriented chopped or swirled filaments loosely held together with a binder.

MATCHED DIE: A mold, in two or more pieces, which is capable of producing parts with two or more dimensionally controlled surfaces.

MATRIX: The essentially homogeneous material in which the fiber system of a composite is imbedded.

MELTING RANGE: Thermoplastics whose makeup includes a distribution of molecular weights will not have a well-defined melting point, but have a melting range.

MICROCRACKING: Microcracks are formed in composites when thermal stresses locally exceed the strength of the matrix. Since most microcracks do not penetrate the reinforcing fibers, microcracks in a cross-plied tape laminate or in a laminate made from cloth prepreg are usually limited to the thickness of a single ply.

MOISTURE CONTENT: The amount of moisture in a material determined under prescribed conditions and expressed as a percentage of the mass of the moist specimen, i.e., the mass of the dry substance plus the moisture present.

MOISTURE EQUILIBRIUM: The condition reached by a sample when it no longer takes up moisture from, or gives up moisture to, the surrounding environment.

MOLD RELEASE AGENT: A lubricant applied to mold surface to facilitate release of the molded article.

MOLD SURFACE: For an autoclave or hydroclaved laminate, the mold surface is the side of the laminate which faced the mold (tool) during cure.

MOLDED EDGE: An edge which is not physically altered after molding for use in final form, and particularly one which does not have fiber ends along its length.

MOLDED NET: Description of a molded part which requires no additional processing to meet dimensional requirements.

MOLDING: The forming of a composite into a prescribed shape by the application of pressure during the cure cycle of the matrix.

N

NDI: Nondestructive inspection. A process or procedure for determining the quality or characteristics of a material, part, or assembly without permanently altering the subject or its properties.

NDT: Nondestructive testing. Broadly considered synonymous with NDI.

NOVOLAC: A phenolic-aldehyde resin which remains permanently thermmoplastic unless a source of methylene or other groups are added.

O

ORTHOTROPIC: Having three mutually perpendicular planes of elastic symmetry.

OUT TIME: The time a prepreg is exposed to ambient temperature; namely, the cumulative amount of time prepreg is out of the freezer. The main effects of out time are to decrease drape and tack of the prepreg while also allowing it to absorb mositure from the air.

OVEN DRY: The condition of a material that has been heated under prescribed conditions of temperature and humidity until there is no further significant change in its mass.

OXIDATION: In carbon/graphite fiber processing, the step of reacting the precursor polymer (rayon, PAN, or pitch) with oxygen, resulting in stabilization of the structure for the hot stretching operation. In general usage, oxidation refers to any chemical reaction in which electrons are transferred.

P

PAN: Polyacrylonitrile, used in fiber form as a percursor for making carbon/graphite fibers.

PAS: Polyarysulfone.

PEEL PLY: A layer of open-weave material, usually fiberglass or heat-set nylon, applied directly to the surface of a prepreg layup. The peel ply is removed from the cured laminate immediately before bonding operations, leaving a clean, resin-rich surface which needs no further preparation for bonding, other than application of a primer where one is required.

PEEK: Polyetheretherketone.

PES: Polyethersulfone.

PHENOLIC: Any of several types of synthetic thermosetting resin obtained by the condensation of phenol or substituted phenols with aldehydes such as formaldehyde.

PI: Polyimide.

PITCH: High molecular weight material left as a residue after processing of petroleum (crude oil). After further purification, can be processed into fiber form, useful as a precursor for production of carbon/graphite fibers.

PITCH FIBER: Fibers derived from a special petroleum pitch.

PLAIN WEAVE: A weaving pattern where the warp and fill fibers alternate; i.e., the repeat pattern is warp/fill/warp...Both faces of a plain weave are identical. Properties are significantly reduced relative to a weaving pattern with fewer crossovers.

PLASTIC: A general term for the mixture of a polymer and ingredients such as hardeners, fillers, reinforcing fibers, plasticizers, etc. After processing, thermoplastics can be resoftened to their original condition by heat, while the thermosetting plastics cannot.

PLASTICIZER: A material of lower molecular weight added to a polymer to separate the molecular chains. This results in a depression of the glass-transition temperatures, reduced stiffness and brittleness, and improved processability.

PLY: A single layer of prepreg. Used synonymously with "Lamina."

PLY WRINKLE: A condition where one or more of the plies are permanently formed into a ridge, depression, or fold.

POISSON'S RATIO: The ratio of transverse strain to the corresponding axial strain below the proportional limit caused by a uniformly distributed axial stress.

POLYAMIDEIMIDE: A polymer containing both amide ("nylon") and imide (as in polyimide) groups; properties combine the benefits and disadvantages of both.

POLYARYLSULFONE (PAS): A high temperature thermoplastic previously marketed under the trade name of Astrel 360. "Polyarylsulfone" is also occasionally used to describe the family of resins which includes polysulfone and polyethersulfone.

POLYIMIDE (PI): Generic name for a family of high temperature resins. Both thermoplastic and thermosetting versions are available.

POLYMER: An organic material composed of long molecular chains consisting of repeating chemical units.

POLYMERIZATION: A chemical reaction in which the molecules of nonomers are linked together to form polymers.

POLYPHENYLENE SULFIDE (PPS): A high temperature thermoplastic useful primarily as a molding compound. Optimum properties depend on slightly cross-linking the resin. Best known under the trade name of Ryton.

POLYPHENYLSULFONE: A relatively new thermoplastic having properties similar to polyethersulfone but with increased resistance to water and some solvents.

POLYSULFONE: A thermoplastic polymer with the sulfone linkage with a T_g of 375°F (190°C).

POROSITY: A condition of trapped pockets of air, gas, or voids within a cured laminate, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material. See Voids..

POSTCURE: Completing the cure cycle of a laminate in an oven instead of tying up the equipment used for the initial cure.

POT LIFE: The period of time during which a reacting thermosetting composition remains suitable for its intended processing after mixing with a reaction-initiating agent.

PPS: Polyphenylene Sulfide. Better known under the trade name of Ryton.

PRECURSOR: In carbon/graphite fiber technology, the organic fiber which is the starting point for making carbon or graphite fibers. In resin technology, sometimes used to describe the polymers present at an intermediate stage in the formulation of a cured resin.

PREFIT: A process to check the fit of mating detail parts in an assembly prior to adhesive bonding to ensure proper bond lines. Mechanically fastened structures are also prefit sometimes to establish shimming requirements.

PREMOLDING: The layup and partial cure at an intermediate cure temperature of a laminated or chopped fiber detail part to stabilize its configuration for handling and assembly with other parts for final cure.

PREPLY: A composite material lamina in the raw material stage ready to be fabricated into a finished laminate. The lamina is usually combined with other raw laminae prior to fabrication. A preply includes all of the fiber system placed in position relative to all or part of the required matrix material that together will comprise the finished lamina. An organic matrix preply is called a prepreg. (Metal matrix preplies include green tape, flame-sprayed tape, and consolidated monolayers.)

PREPREG: Ready to mold or cure material in sheet form which may be fiber cloth, or mat, impregnated with resin and stored for use. The resin is partially cured to a "B" stage and supplied to the fabricator for layup and cure.

PRESS CLAVE: A simulated autoclave made by using the platens of a press to seal the ends of an open chamber, providing both the force required to prevent loss of the pressurizing medium and also providing the heat to cure the laminate inside.

PRESSURE INTENSIFIER: A layer of flexible material (usually a high temperature rubber) used to assure that sufficient pressure is applied to a location, such as a radius, in a layup being cured.

PROCESS CONTROL: During laminate cure, the use of electrical technique to monitor the cure cycle. Also refers to the overall procedure of recorging cure cycle temperatures (in or near the part), vacuum, and pressure, plus mechanical testing of a "process control" panel cured along with the part.

PULTRUSION: A process to continuously process structural shapes or flat sheet by drawing prepreg materials through forming dies to produce the desired constant cross-sectional shape and simultaneously curing the resin.

Q

QUASI-ISOTROPIC: A layup sequence of the 0°, +45°, −45°, 90° family, with equal amounts of fiber in each direction. With the fiber axes in four directions, laminate properties in the plane of the fibers are nearly isotropic.

R

REINFORCEMENT: A relatively high strength or stiffness material inbedded in a matrix to improve their mechanical properties.

RELEASE AGENT: See Mold Release Agent.

RELEASE FILM: An impermeable layer of film which does not bond to the resin being cured. See Separator.

RESIN: A polymer (or polymers) and their associated hardeners, catalysts, acclerators, etc., which can be converted to a solid by application of energy, normally in the form of an elevated temperature.

RESIN CONTENT: The amoung of matrix present in a composite by percent weight.

RESIN RICHNESS: An area of excess resin, usually occurring at radii, steps, and the chambered edge of core.

RESIN STARVED: An area deficient in resin usually characterized by excess voids and/or loose fibers.

ROVING: A number of strands, tows or ends collected into a parallel bundle with little or no twist.

RUBBER: Crosslinked polymers whose glass transition temperature is below room temperature and which exhibit highly elastic deformation and have high elongation.

S

SANDWICH CONSTRUCTION: A structural panel consisting in its simplest form of two relatively thin, parallel sheets of structural material bonded to and separated by a relatively thick, lightweight core.

SCRIM: (also called Glass Cloth, Carrier): A reinforcing fabric woven into an open mesh construction, used in the processing of tape or other B-stage material to facilitate handling.

SECONDARY BONDING: The joining together, by the process of adhesive bonding, of two or more already cured composite parts.

SEMICRYSTALLINE: In plastics, refers to materials which exhibit localized crystallinity. See Crystallinity.

SEPARATOR: A permeable layer which also acts as a release film. Porous Teflon coated fiberglass is an example. Often placed between layup and bleeder to facilitate bleeder system removal from laminate after cure.

SHELF LIFE: The length of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and/or remain suitable for its intended function.

SHELL TOOLING: A mold or bonding fixture consisting of a contoured surface shell supported by a substructure to provide dimensional stability.

SHORT BEAM SHEAR: A flexural test of a specimen having a low test span to thickness ratio (e.g., 4/1) such that failure is primarily in shear.

SIZING: Material applied as a very thin coating on fibers to improve their processability and/or to increase the fiber/matrix bond strength in composites. See Fiber Finish.

SPECIFIC GRAVITY: The weight of a specimen compared to the weight of the volume of water it displaces.

STAGING: Heating a pre-mixed resin system, such as in a prepreg, until the chemical reaction (curing) starts, but stopping the reaction before the gel point is reached. Stating is often used to reduce resin flow in subsequent press molding operations.

STOPS: Metal pieces inserted between die halves; used to control the thickness of a press molded part. Not a recommended practice, since the resin will end up with less pressure on it, and voids can result.

STRESS, RESIDUAL: The stress existing in a body at rest, in equilibrium, at uniform temperature an not subjected to external forces.

SYMMETRICAL LAMINATE: A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

T

TACK: Stickiness of a prepreg.

TACKING: To locally join together layers of thermoplastics, by localized melting of the resin.

TAPE: In composites technology, tape is the prepreg form consisting of collimated fibers and resin, supported on a layer of release paper. Almost all graphite tape is made to have an 0.005″ thickness, as cured.

TEFLON: DuPont trade name for both TFE and FEP flourocarbon polymers.

TFE: Tetraflouroethylene, or polytetraflourethylene (PTEE), the "Teflon" with the highest elevated temperature resistance. However, yield strength is very low; Teflon is known for having "cold flow" problems.

THERMAL CONDUCTIVITY: The capability of a substance to "conduct" heat from a hot area to a cooler area. Graphite fibers have good conductivity (both thermal and electrical) in the axial direction, but relatively poor conductivity in the radial direction.

THERMOFORMING: Forming a thermoplastic material after heating it to the point where it is soft enough to be formed without cracking or breaking reinforcing fibers.

THERMOPLASTIC: A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and that in the softened stage can be shaped by flow into articles by molding or extrusion.

THERMOSET: A plastic that is substantially infusible and insoluble after having been cured by heat or other means.

TOUGHNESS: Describes a material which has both high elongation to failure and good strength, such that the area under the stress/strain curve (a measure of the energy required to deform the material) is very high.

TOW: An untwisted bundle of continuous filaments. Commonly used in referring to man-made fibers, particularly carbon and graphite fibers in the composites industry.

TRACER: A fiber or tow or yarn added to a pregreg for verifying fiber alignment and, in the case of woven materials, distinguishing warp fibers from fill fibers.

V

VACUUM BAG: The plastic or rubber layer used to cover the part so that a vacuum can be drawn.

VACUUM BAG MOLDING: A process in which the layup is cured under pressure generated by drawing a vacuum in the space between the layup and a flexible sheet placed over it and sealed at the edges.

VENT CLOTH: A layer or layers of open weave cloth used to provide a path for vacuum to "reach" the area over a laminate being cured, such that volatiles and air can be removed and also causing the pressure differential that results in application of pressure to the part being cured. Often used synonymously with "Breather Cloth".

VENTING: In autoclave curing a part of assembly, venting refers to turning off the vacuum source and venting the vacuum bag to the atmosphere. The pressure on the part then becomes the pressure difference between the pressure in the clave and atmospheric pressure. Venting is usually used to prevent the resin "boiling" that can occur when a resin is heated and simultaneously subjected to reduced pressure (vacuum).

VISCOSITY: The property of resistance to flow exhibited within the body of a material.

VOID: A physical and mechanical discontinuity occurring within a material or part which may be 2-D (e.g., disbonds, delaminations) or 3-D (e.g., vacuum-, air-, or gas-filled pockets). Porosity is an aggregation of micro-voids. Voids are essentially incapable of transmitting structural stresses or nonradiative energy fields.

VOLATILES: Refers to gaseous materials leaving a laminate that is being cured, and which were liquids or solids before the cure cycle started. Volatiles produced usually include residual solvents and absorbed or adsorbed water. Many materials also produce volatiles as by-products of the curing reactions.

W

WARP: The longitudinally oriented yarn in a woven fabric (see Fill); a group of yarns in long lengths and approximately parallel.

WATER JET: Water emitted from a nozzle under very high pressure. Useful for cutting materials.

WET LAYUP: A method of making a reinforced product by applying the resin system as a liquid as the reinforcement is put in place.

WET STRENGTH: The strength of a composite measured after conditioning the test specimen in water or water vapor.

WET WINDING: A method of making filament in which the fiber reinforcement is coated with the resin system as a liquid just prior to wrapping on a mandrel.

WETOUT: The process of wetting a fiber bundle with a resin; that is, wetout is achieved when all of the fiber surface area is in intimate contact with the matrix resin.

WHISKER: A short single fiber or filament. Whisker diameters range from 1 to 25 microns with length-to-diameter ratios between 100 and 15,000.

WORK LIFE: The period during which a compound, after mixing with a catalyst, solvent, or other compounding ingredients, remains suitable for its intended use.

Y

YARN: Generic term for strands of fibers or filaments in a form suitable for weaving or otherwise intertwining to form a fabric.

Z

ZERO BLEED: A laminate fabrication procedure which does not allow loss of resin during cure. Also describes prepreg made with the amount of resin desired in the final part, such that no resin has to be removed during cure.

PREFACE

In the past decades considerable progress in advanced composite technology has been made. However, the full potential in the design, manufacturing and especially the application of composites has not been realized. The use of composites in heavily loaded primary structures has been limited, mainly due to lack of hands-on experience and confidence. This book is intended to advance the technical understanding and practical knowledge of advanced composites, emphasizing the design and manufacture of airframe structures. All aspects at composite design will be discussed in a thorough and rigorous fashion which includes guidelines, observations, design factors, pros and cons of design cases, and troubleshooting techiques. However, neither the basic chemistry of materials nor laminate strength (or stress) analysis will be discussed in detail. Such information can be found in numerous composite books and published papers.

This book (which may be used as a handbook) is divided into twelve chapters, with emphasis on itemized write-ups, tables, graphs and illustrations which focus on points of interest. While some of the data and information will be superseded as technology advances, the basic technical information and data will hold true. However, some modification of the design concepts and manufacturing processes described in this book may be required in due course. One purpose of this book is to give composite designers and engineers a practical design tool, containing broad data and information gained from past experience and lessons learned in design and fabrication of composite components, that can be used to design low cost and weight efficient composite structures with high structural integrity.

Composite structures are not just an extension of their metal counterparts and should not be considered only as piece-by-piece replacements (the BLACK ALUMINUM approach) that merely save structural weight by their lower material density advantage. The designer's ingenuity and resourcefulness is needed to develop innovative concepts which will reach the ultimate goal of composite structures that meet the requirements of durability, damage tolerance, maintainability, repairability, crashworthiness, low weight and cost effectiveness.

Early interface and support from producibility at the predesign level is critical in composite design to insure that cost-effective producibility features. Engineering design should also seek interface and criteria from tool design, production, manufacturing, industrial engineering, and quality assurance. It is recommended that during the composite design process on-board support guidelines are combined with previous counterpart metal experience. It must be remembered that composite airframe structural design encompasses almost all the engineering disciplines, and engineers who go to the computer workstation or drawing board today need hands-on information and data that are technically sound and emphasize rational design and technical analysis, and this book was written with that need in mind. It is not practical, however, to cover all the information and data within this handy compact book. Selected relevant references are presented at the back of each chapter so that reader can explore his own personal interests in greater detail.

In preparing this book, it was necessary to collect vast amounts of information and data from many sources. Sincere appreciation and thanks is given to those who contributed to this book, to my previous colleagues at Lockheed Aeronautical Systems Company, and to other specialists from various companies for their gracious help. Special thanks to **Dr. John T. Quinlivan**, Manager of B777 Empennage Structures, The Boeing Company, for his valuable comments and review of the entire draft of this book.

While some of the material presented in this book may be controversial, the author has tried to the best of his knowledge to make this book of practical use in the airframe structures arena. Any constructive suggestions and comments for improvement and future revision would be greatly appreciated by the author.

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L.A., California, U.S.A.

Jan., 1992

PREFACE – THIRD EDITION

This third edition still has twelve chapters and there is no change for those chapters, except the addition of a section, 12.7 Green Airframe, at the end of Chapter 12.0. However, in the back of this book three additional appendices are added for the reader's information, i.e., Appendix D – Review of Certification for Composite Airframe, Appendix E – Brief Summary of the B787, and Appendix F – Brief Summary of the A350XWB.

Appendix D compiles many relevant documents and information for certifying composite airframe structures. Airworthiness standards of FAR23.573(a) are used for certifying the composite structures of normal, utility, acrobatic, and commuter category airplanes. FAA advisory circular AC20-107B (published in September, 2009) is for compliance with certification requirements of civil composite aircraft structures. This AC is also important in certification of civil transport airframe from FAR 25, which is the airworthiness standards for transport category airplanes.

Appendix E introduces the Boeing B787, which has used more than 50% composite material for its airframe. B787 also uses advanced technology to manufacture composite fuselage, wing, and horizontal tail main boxes. The fuselage uses the tape-winding method to make several single-barrels with varying thickness of solid laminate skin. Eventually, all these barrels are circumferentially joined together by mechanical fasteners. The wing box consists of one piece laminate skin bonded together with I-stringers to form both upper and lower panels. The horizontal tail box is a co-cured multi-spar construction, which is different than the conventional multi-rib design, for the big transport tail box.

Appendix F introduces the Airbus A350XWB (XWB means Extra-Wide Body) which is 8 inches wider than the B787. Its airframe also uses more than 50% composite materials to reduce structural weight. However, its fuselage barrel skins are fabricated with four panels (two side panels, one upper panel and one lower panel) and they are joined together by conventional mechanical fasteners. Those panels are also bolted on the aluminum / lithium frames. The wing box consists of two composite spars and cover panels, combining with metallic machined aluminum / lithium ribs.

All data in these three appendices are merely for the reader's information and reference only. However, the reader should refer to latest data available from airplane certifying agencies and / or the airplane manufacturing companies.

Sincere appreciation and thanks to Professor Xiaoquan Cheng, Beijing University of Aeronautics and Astronautics, for his review and valuable comments for the entire third edition.

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Los Angeles, California, U.S.A.

December, 2010

CONTENTS

Chapter 2.0

MATERIALS

2.1 INTRODUCTION

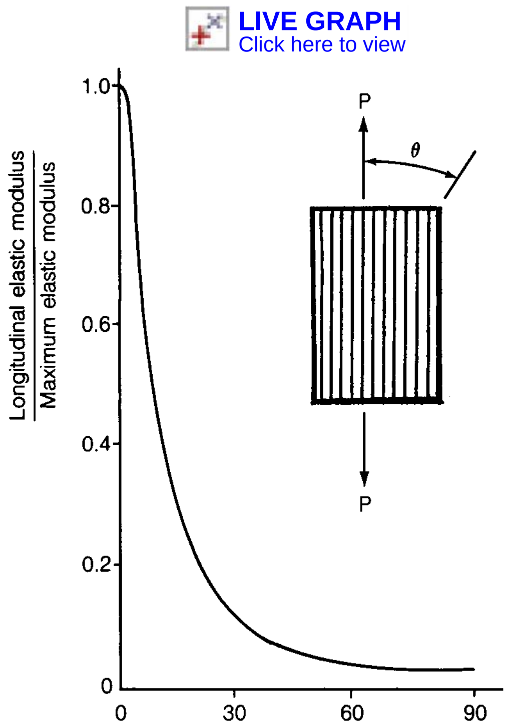
Composite materials have gained their acceptance among structural engineers during the last decades. The performance of a composite depends upon:

- The composition, orientation, length and shape of the fibers;
- The properties of the material used for the matrix (or resin);
- The quality of the bond between the fibers and the matrix material.

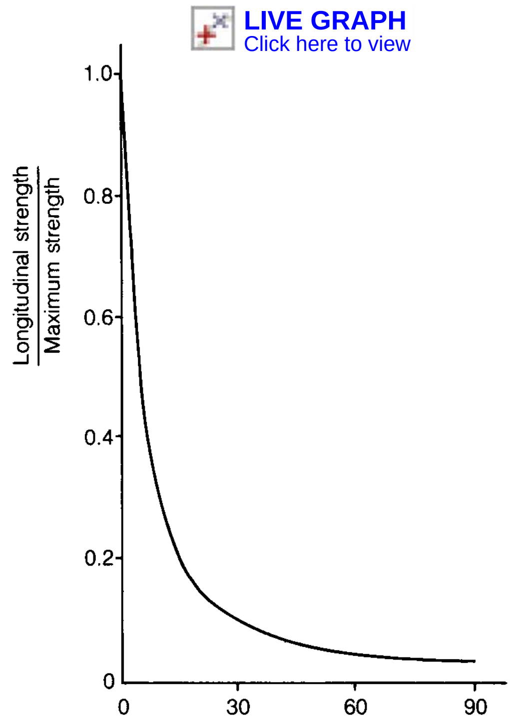
Composite materials consist of any of various fibrous reinforcements coupled with a compatible matrix to achieve superior structural performance. The most important contribution to material strength is that of fiber orientation. Fibers can be unidirectional, crossed ply, or random in their arrangement and, in any one direction, the mechanical properties will be proportional to the amount of fibers oriented in that direction. Reduced properties result from the shear strength of the weak matrix. In fact, both the strength and moduli of a composite in a ply are reduced considerably when the angle of the applied load deviates from the direction of the filaments in the composites. Fig. 2.1.1 shows how a unidirectionally reinforced composite will have a far lower strength in transverse tension than one loaded exactly in the direction of the fibers. Therefore, it is evident that with increasingly random directionality of fibers, mechanical properties in any one direction are lowered. Thus, because of their low mechanical properties normal to the fiber direction, laminate composites will need to be strengthened or stiffened by laying up plies (unidirectional tape or woven fabric) in different directions. Such lamination will be necessary because stresses in a loaded component or panel can vary in both the "X" and "Y" direction.

Laminate properties of various combinations of plies oriented at different angles can be calculated through the use of computer programs to produce the best design. These computer aids are particularly helpful, because of the composite's non-isotropic properties, in calculating the various properties of any combination of oriented plies.

In this Chapter only those materials which are used on airframe structures will be discussed the data given is general and, while it may be used by the designer to do rough sizing, it is not appropriate for stress analysis or final sizing use. No composite material design allowable data is given (usually this data is part of a company's proprietary data) in this chapter because numerous varieties exist and many improved products are available every year.



(a) Modulus



(b) Strength

Fig. 2.1.1. Modulus And Strength Of Composites Drop Steeply As The Angle Between The Fibers And The Direction Of Load Increases

(1) MATERIAL SELECTION

Material selection plays a large part in final cost, not only because the raw material itself is expensive but also because the material selected often determines downstream manufacturing costs. The material selection criteria are given below:

- Cost
- Available mechanical and environmental properties database
- Suitability for use in proposed manufacturing processes
- Structural performance
- Ease of processing
- Ease of handling
- Supportability
- Maximization of knowledge base
- Available processing data
- Immediate or near term commercial availability

There are many cases where more than one material can meet the structural and/or weight requirements specified for a given part. Assume there is a choice between a unidirectional material form (which can be used on the automated lay-up machine) and a broadgoods form (fabric or woven) of the same material. Clearly there is a difference in the costs of these raw material forms: unidirectional prepreg generally less expensive because the material supplier has not gone through the added step of weaving the broadgoods fabric. At the same time, it may take more time to fabricate a laminate component from unidirectional tape than form broadgoods. Therefore, there is a tradeoff between actual raw material costs and the downstreams manufacturing costs which are predetermined in choosing a particular raw material.

Obviously, along with cost considerations, any design must carefully match the requisite properties with the candidate composite material. Once the optimum, or best available, material is chosen, the designer must be concerned with any additional limitations that material selection might impose on the capabilities of the design. The common areas of concern include hot/wet properties, notched effect (if fasteners or small cutouts are used in design), in-service temperature, and impact resistance. Critical limitations that must be considered in composite design include the relatively low strength and stiffness in the out-of-plane direction and often poor shear properties. These factors, must be considered to prevent delamination under compressive loading or inadequate out-of-plane load-carrying capabilities.

Various kinds of composite materials with temperature resistant matrices and high-performance reinforcements are currently available or are in advanced stages of development. Their upper (and partially overlapping) service temperatures are shown in Fig. 2.1.2.

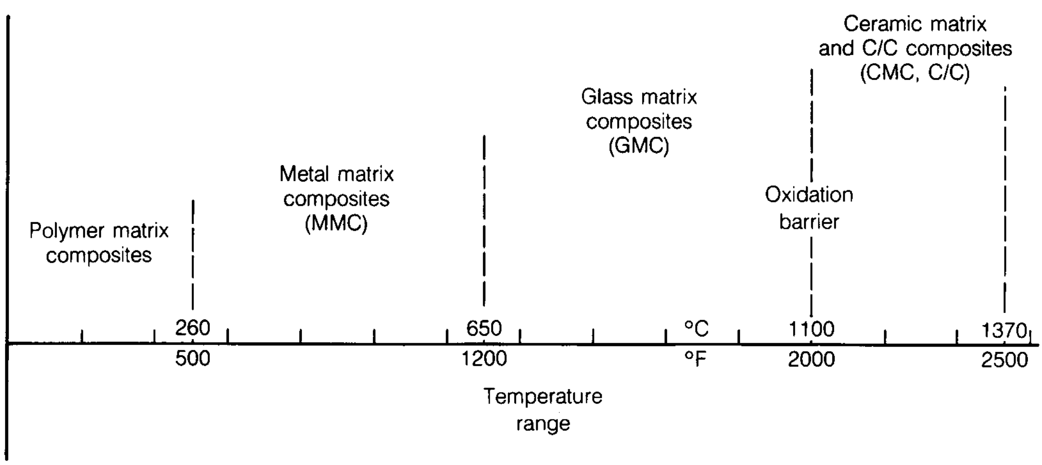


Fig. 2.1.2 Temperature Resistance of Composites

Since the properties of composites depend critically on the processes used to make them, designers must work with prepreg and fiber producers to achieve desired results. The designers should be aware of the weaknesses of various fibers and construction methods and so design around them. The following are material specification requirements:

- Fiber and fabric properties
- Storage and retest requirements
- Uncured prepreg properties
- Cured or co-consolidated prepreg properties
- Mechanical properties
- Environmental testing
- Processability trials
- Chemical characterization
- Non-destructive inspection (NDI)

In general, most of the major reinforcement systems have been well characterized for many years, and performance improvements have occurred in relatively small increments. However, improvements in the matrix resins (both thermoset and thermoplastic) have allowed great strides in composite fabrication, producibility, performance, and stability.

The reinforcing fiber may have a negative thermal expansion coefficient along its axis, a property that makes possible the design of structures with zero or very low linear and planar thermal expansion. Thus, the main support truss for the mirrors of the "Space Telescope" is made of a carbon/epoxy composite to meet extreme close tolerance requirements.

It is worthwhile to note research on organic conducting polymers, which would have many airframe applications, such as to provide shielding on composite structures for sensitive control electronics from electro-magnetic interference (EMI). Another related application is lightning strike tolerance on airframe structures.

(2) HYBRID LAMINATES

Hybrid systems, made by combining two or more types of fibers in a single laminate, can be tailored to meet specific performance requirements, and are an effective means of reducing the cost of composites. The unique performance of one of the reinforcing materials compared to the other can enable the composite to do a job that neither can do independently. While hybrid composites any offer the best choices for some design cases, designing with hybrids is somewhat complicated because most of their properties are not as easily characterized as are those of single fiber composites. Applications and data show that different fibers can be combined successfully in a structure in many ways. The fibers can be used in different layers or even in completely different parts of the same structural element. They also can be blended to form a hybrid tape or woven to form a hybrid fabric.

For hybrid constructions, directional response and failure parameters should be defined for each material. Care must be taken to provide reinforcement for all loading directions. Since carbon or graphite is conductive both thermally and electrically and has a slightly negative coefficient of thermal expansions, it is conceivable for designers to develop hybrid material geometries with structural responses totally different from the existing conventional metal materials.

As the cost/performance tradeoff becomes more critical, hybrids may become the material system of choice for more structural uses, making them materials with a future. Nevertheless, the future of hybrids in the airframe industry appears uncertain and much still needs to be learned about this systems. Issures peculiar to hybrid systems are described below:

(a) Material more tailored to specific needs than is available with a single fiber-matrix combination is very desirable.
(b) Different fibers have different:
- Strains to failure
- Moduli
- Coefficients of thermal expansion
- Coefficients of moisture expansion

(c) Thermally induced stresses exist in every hybrid lamina (below the laminate level)
- Caused by different thermal expansion characteristics of constituents
- Can be large enough to cause failure without mechanical load

(d) Effectively an infinite variety of hybrids is possible
- Each new hybrid must have some minimum level of material property qualification
- The wide variety of possible hybrids (just like the wide variety of possible laminates) must be deliberately restricted on purely practical grounds.

There are typically three methods of hybridization:
- Interply — Different reinforcements are stacked in separate layers with no mixing within the layers
- Intraply — Different reinforcements are commingled within a layer either by alternating strands or mixing chopped fibers
- Selective placement — The laminate is basically composed of one reinforcement, but a different reinforcement is added in certain areas (such as corners, ribs, etc.)

Hybrid reinforcements can be combined in almost all material forms including:
- Prepregs
- Fabric
- Woven roving
- Chopped fibers

Common hybrids include:

(a) Carbon/Aramid
Can be combined without residual thermal stresses since coefficients of thermal expansion are very similar

(b) Carbon/Glass
- Increased impact strength
- Improved fracture toughness
- No galvanic corrosion
- Reduced cost over all carbon fiber laminates

(3) MATERIAL PROPERTIES SUMMARY

A summary of material density, coefficient of thermal expansion and thermal conductivity data is shown in Fig. 2.1.3.

Material	Density lb/in^3	Thermal Expansion in/in/°F$\times 10^{-6}$	Thermal Conductivity (BTU) (in.) / (hr) (ft^2) (°F)
Aluminum	0.1	12.6-13	1300-1400
Beryllium	0.066	6	1121
Magnesium	0.063	14	1092
Steel	0.284	6-6.7	310-460
Titanium	0.160	4.7-5.6	112
Glass	0.09	3.0	5.4
Phenolic	0.048	11-35	1.1
Epoxy	0.0457	20-60	2-6
Polyester	0.046	16-35	2-5
Polyimide	0.052	70	3-5
Silicone rubber	0.045	45-200	2
Kevlar 49	0.052	−3.5	—
Silicon	0.11	1.4	—
Boron	0.098	2.7-3.5	22
Carbide	0.125	2.2	—
Ceramic	0.058-0.072	0.5-5	10-80
Monolithic Graphite	0.063	1-2	800
Boron filament	0.083	2.7	—
Graphite fiber	0.063	−0.05	840
Invar	0.29	3	—
Graphite/epoxy [0]	0.055-0.059	0.24	—
E Glass/epoxy [0]	0.065-0.070	4.8	—
Kevlar 49/epoxy [0]	0.046-0.050	−3.0	—
Boron/epoxy [0]	0.07-0.075	2.3	—
Nickel	0.32	7.4	400

Fig 2.1.3 Summary of Material Density, Coefficient of Thermal Expansion and Thermal Conductivity Data

2.2 ORGANIC MATRICES

The purpose of the matrix is to bind the reinforcement (fiber) together and to transfer load to and between fibers, and to protect the flaw- or notch-sensitive fibers from self-abrasion and externally induced scratches. The matrix also protects the fibers from environmental moisture and chemical corrosion or oxidation, which can lead to embrittlement and premature failure. In addition, the matrix provides many essential functions from an engineering standpoint: the matrix keeps the reinforcing fibers in the proper orientation and position so that they can carry the intended loads, distributes the loads more or less evenly among the fibers, provides resistance to crack propagation and damage, and provides all of the interlaminar shear strength of the composite. The matrix generally determines the overall service temperature limitations of the composite and may also control its environmental resistance.

In summary, the matrix:

- distributes loads through the laminate
- protects fibers from abrasion and impact
- determines:
  - — compressive strength
  - — transverse mechanical properties
  - — interlaminar shear
  - — service operating temperature
  - — selection of fabrication process and tool design
- contributes to fracture toughness

With any fiber, the material used for the matrix must be chemically compatible with the fibers and should have complementary mechanical properties. Also, for practical reasons, the matrix material should be reasonably easy to process.

The development of high strength and high thermal resistance is frequently accompanied by complex cure procedures or brittleness in thermosets. Overcoming these obstacles has proven the key to developing viable composite matrices, with processing/fabrication constraints of fiber wet-out, prepreg shelf life, tack and drape, cure shrinkage, etc., adding to the complexity.

The organic matrices commonly used are broadly divided into the categories of thermoset and thermoplastic; organic matrices commonly used on airframe structures are given below:

THERMOSET	THERMOPLASTIC
• Expoxy	• Polyethylene
• Polyester	• Polystyrene
• Phenolics	• Polypropylene
• Bismaleimide (BMI)	• Polyetheretherketone (PEEK)
• Polyimides	• Polyetherimide (PEI)
	• Polyethersulfone (PES)
	• Polyphenylene Sulfide
	• Polyamide-imide (PAI)

The relative advantages of thermosets and thermoplastics include:

THERMOSET MATRICES	THERMOPLASTIC MATRICES
(Characteristics)	
• Undergo chemical change when cured	• Non-reacting, no cure required
• Processing is irreversible	• Post-formable, can be reprocessed
• Low viscosity/high flow	• High viscosity/low flow
• Long (2 hours) cure	• Short processing times possible
• Tacky prepreg	• Boardy prepreg
(Advantages)	
• Relatively low processing temperature	• Superior toughness to thermosets
• Good fiber wetting	• Reusable scrap
• Formable into complex shapes	• Rejected parts reformable
• Low viscosity	• Rapid (low cost) processing
	• Infinite shelf life without refrigeration
	• High delamination resistance
(Disadvantages)	
• Long processing time	• Less chemical solvent resistance than thermosets
• Restricted storage	• Requires very high processing temperatures
• Requires refrigeration	• Outgassing contamination
	• Limited processing experience available
	• Less of a database compared to thermoset

Compared to thermoplastics, thermoset matrices offer lower melt viscosities, lower processing temperatures and pressures, are more easily prepregged and are lower cost. On the other hand, thermoplastic matrices offer indefinite shelf life, faster processing cycles, simple fabrication, and generally do not require controlled-environment storage or post curing.

(1) THERMOSET MATRICES

The most prominent matrices are epoxy, polyimides, polyester and phenolics; a matrix comparison is given in Fig. 2.2.1.

Thermoset matrix systems have been dominating the composite industry because of their reactive nature. These matrices allow ready impregnation of fibers, their malleability permits manufacture of complex forms, and they provide a means of achieving high-strength, high-stiffness crosslinked networks in a cured part. Fig. 2.2.2 is a comparison of selected thermoset matrices which are commonly used in airframe design for primary structural applications.

Characteristics	Thermosetting resins				
Property	Polyester	Epoxy	Phenolic	Bismaleimide	Polyimide
Processability	Good	Good	Fair	Good	Fair to difficult
Mechanical properties	Fair	Excellent	Fair	Good	Good
Heat resistance	180°F	200°F	350°F	350°F	500-600°F
Price range	Low — Medium	Low — Medium	Low — Medium	Low — Medium	High
Delamination resistance	Fair	Good	Good	Good	Good
Toughness	Poor	Fair — Good	Poor	Fair	Fair
Remarks	Used in secondary structures, cabin interiors, primarily with fiberglass	Most widely used, best properties for primary structions; principal resin type in current graphite production use	Used in secondary structures, primarily fiberglass good for cabin interiors for low smoke generation	Good structural properties, intermediate temperature resistant alternative to epoxy	Specialty use for high temperature application

Fig. 2.2.1 Comparison of Properties for General Thermoset Matrices

Matrix Type	Tack	Drape	Thermal Stability	Cure Temp	Cure Pressure	Void Content	Cost	Other Problems
Epoxy	Excellent	Excellent	200°F Dry 180°F Wet	350°F (177°C)	100 psi	Low	Low	Low temp Storage High Moisture Pickup Brittle
Toughened Epoxy	Very good	Excellent	180°F Dry 160°F Wet	350°F (177°C)	100 psi	Low	Moderate	Low temp Storage
BMI	Good When heated	Good When heated	350°F to 400°F Dry 300-350°F Wet	350°F Cure 400 — 500°F Post cure	100 psi	Low	Moderate	Microcracks on tempertaure Cycling
Condensation PI	Good	Good	Excellent 500°-600°F	600°-700°F	500 psi or higher	High	Moderate	Complex process cycle Large parts difficult
Acetylene Terminated PI	Poor-boardy	Poor	Very good 500-550°F	500°F Cure 600°F — 650°F post	200 psi	Low	Moderate to high	Small process window
PI (PMR)	Good	Good	400-500°F Higher for brief periods	Complex to 650°F	500 psi	Low to moderate	Moderate	

Fig. 2.2.2 Comparison of Commonly Used Thermoset Matrices

(A) EPOXY

Epoxy systems are the major composite material for low-temperature applications [usually under 200°F (93°C)] and generally provide outstanding chemical resistance, superior adhesion to fibers, superior dimensional stability, good hot/wet performance, and high dielectric properties. Epoxy can be formulated to a wide range of viscosities for different fabrication processes and cure schedules. They are free from void-forming volatiles, have long shelf lives, provide relatively low cure shrinkage, and are available in many thoroughly-characterized standard prepreg forms. They also have good chemical stability and flow properties, and exhibit excellent adherence and water resistance, low shrinkage during cure, freedom from gas formation, and stability under environmental extremes. In addition, on other very important advantage is the wealth of database information available.

The epoxy family is the most widely used matrix system in the advanced composites field. Because it is generally limited to service temperatures, this restricts use in many aerospace applications, where higher service temperatures are required.

The baseline system of epoxy used in the majority of applications includes:

- Superior mechanical properties
- 250°F curing: — 65°F to 180°F (−53 to 82°C) service temperatures
- 350°F curing: — 65°F to 250°F (−53 to 121°C) short-term or 200°F (93°C) long-term service temperature

Epoxy matrices, the workhorse of the advanced composites industry today, are suitable for use with glass, carbon/graphite, aramid, boron, and other reinforcements and hybrids. Yet greater demands can be met by conventional epoxies are being made for today's parts, so a wide variety of epoxies are being developed to handle the ever-increasing requirements for speed of fabrication, toughness, and higher service temperatures.

Unmodified epoxies are brittle. When subjected to impact from a flying stone, an occasional bump, or a dropped wrench, etc., they can be damaged internally and suffer loss of laminate compressive strength. Epoxies have been modified or improved to increase their damage resistance. The result is "toughened" epoxies.

Epoxies have a tendency to absorb moisture, this absorbed moisture can lead to decrease mechanical properties especially at elevated temperatures. The presence of water decreses the glass transition temperature of the epoxy matrix, hence the term "wet Tg" This effect must be considered in design.

The following environmental hazards have detrimental effects on epoxy matrices:

- Moisture
- Temperature
- Ultraviolet light
- Hydraulic fluid
- Fuel
- Cleaning agents

(B) POLYIMIDES (HIGHER SERVICE TEMPERATURE MATRICES)

Polyimides are thermo-oxidatively stable and retain a high degree of their mechanical properties at temperatures far beyond the degradation temperature of many polymers, often above 600°F (320°C). Several types with superior elevated temperature resistance are listed below:

- Bismaleimides: good to 450°F (230°C), relatively easy to process
- Condensation types: good to 600°F (320°C), very difficult to process
- Addition types: good to 500 — 600°F (260 — 320°C), improved processability compared to condensation types

(a) BMI (Bismaleimides), a special polyimide system, operates around a 350°F to 450°F (177 to 230°C) upper limit. BMI offer good mechanical strength and stiffness, but are generally brittle and may have cure-shrinkage. Other BMIs have significant improvements in toughening which greatly enhances their usefulness. When good hot/wet performance or thermal stability beyond epoxy limits is desired, BMI matrices may be the matrix of choice. BMI characteristics are summarized below:
  - BMIs provide increased thermal stability compared to epoxies, with comparable processability
  - The major problem with BMIs has been increased brittleness over epoxies — with reduced damage resistance and toughness
  - BMI systems with improved toughness are available at the sacrifice mechanical properties

(b) PMR-15 (Polymerization of Monomeric Reactants) is a thermoset addition polyimide which offers higher continuous service temperatures. Originally developed by NASA. Thermo-oxidative stability, relatively low cost, and availability in a variety of forms make PMR-15 one of the candidates for airframe industrial applications where performance from 500 to 600°F (260° — 320°C) is the key material selection criterion. PMR-15 processing is complicated, requiring application of near 600 psi (4.1 Mpa). A heated tool is often necessary to achieve faster heatup rates than are possible with conventional tooling in an autoclave. The room temperature properties of PMR-15 are similar to those of 350°F (177°C) epoxy, but, unlike epoxy, properties do not decrease significantly until temperatures over 500°F (260°C) are reached, even after exposure to moisture. NASA has developed LARC-160, a "PMR" system which provides a significant improvement in processability over the PMR-15 matrix, with only a small loss in elevated temperature properties. However, both NASA's PMR-15 and LARC-160 matrices are still in progress under their continuing development programs.

(C) POLYESTER

Polyesters matrices can be cured at room temperature and atmospheric pressure, or at a temperature up to 350°F (177°C) and under higher pressure. These matrices offer a balance of low cost and ease of handling, along with good mechanical and electrical properties, good chemical resistance properties (especially to acids), and dimensional stability. Polyester combined with fiberglass fibers becomes a very good radar-transparent structural material and polyester is also a relatively inexpensive matrix that offers a compromise between strength and impact resistance for use in aircraft radomes. Low mold-pressure requirements helped promote the manufacture of large polyester composite structures, and this was further helped along by their relatively quick cure characteristics.

Vinyl esters are a subfamily of polyesters, derived from epoxy-matrix backbones, which provide higher tensile elongation, toughness, heat resistance, and chemical resistance than conventional unsaturated polyesters.

(D) PHENOLICS

Phenolics are the oldest of the thermoset matrices, and have excellent insulating properties, resistance to moisture, and good electrical properties (except arc resistance). The chemical resistance of phenolics is good, except to strong acids and alkalis. Phenolics are available as compression-molding compounds, and injection-molding compounds. This material is very useful in military and high-performance aerospace applications where radiation-hardness, dimensional stability at high loads and temperatures, and ability to ablate may be critical to component survival.

Fig. 2.2.3 shows a number of the thermoset resin products which are available for reference.

Matrix resin (Vendor)	Resin type	Available Material Forms	Thermal Stability	Remarks
934 (Fiberate) 5208 (Narmco) 3501-6 (Hercules)	Epoxy	(Carbon, glass Kevlar, boron, & ceramic fibers) Unitape Various fabrics Prepreg tow	180°F Serv temp 250°F Dry 180°F Wet	• Best overall structural characteristics • High compressive strength but poor damage tolerance • Absorbs mositure • Structural properties affected by moisture content • Easy to process • Good chemical resistance • Good environmental resistance • Brittle
3502 (Hercules) 8551-7A (Hercules) 8552 (Hercules) R6376 (Ciba Geigy) 977-2 (Fiberite)	Toughened Epoxy	(Carbon & Glass fibers) Unitape Various fabrics	>200°F Dry 180°F Wet	• Same as epoxy but not as brittle because of improved toughness • Lower hot/wet performance than standard epoxy
F650 (Hexcel) 5250-2 (BASF/Narmco) 5245C (BASF/Narmco)	Bismaleimide (BMI)	(Carbon, glass quartz & ceramic fibers) Unitape Various fabrics prepreg tow	250-450°F Dry 250-300°F Wet	• Easy to process • More brittle than epoxies • Microcracks on temperature cycling with some systems
V398 (U.S. poly) F655 (Hexcel) 5250-4 (BASF/Narmco)	Toughened Bismaleimide	Carbon, glass quartz & ceramic fibers) Unitape various fabrics prepreg tow	250-400°F Dry 200-350°F Wet	• Better damage tolerance • Lower service temperature than other BMI's
PMR-15 (Various)	Polyimide	Unitape fabric powder impregnated tow & tape	400-500°F Higher for brief periods	• Difficult to process

Fig. 2.2.3 Thermoset Prepreg Resin Choices (Epoxy and Polyimide)

(2) THERMOPLASTIC MATRICES

In recent years, thermoplastic matrix systems have been introduced. Their major advantages are

- Service temperatures of up to 540°F (280°C)
- Excellent strain capabilities
- High moisture resistance
- Unlimited shelf-life
- Short processing cycles

Disadvantages are

- High processing temperatures
- As yet marginal processing experience
- Lack of drapeability

Thermoplastic matrices are not new to the airframe industry. They have been used for many years for various components, mainly in aircraft fuselage interiors and for other non-structural parts. The engineering thermoplastic resins have high continuous service temperatures, from 250°F to 400°F (121 to 200°C), high matrix melting temperatures, and high viscosity which leads to higher mold pressure in autoclave operations. Thermoplastic matrices provide better interlaminar fracture toughness combined with acceptable postimpact compression, better resistance to high temperatures and solvents, and have low moisture sensitivity. The major advantage over thermoset matrices is their shorter fabrication cycle and the fact that a chemical cure does not take place, allowing reprocessing or reconsolidation of a flawed part after manufacture.

Thermoplastics offer potential cost reduction by:

- Reforming capability
- Welding capability
- Eliminating cold refrigeration storage and having unlimited shelf life
- Rapid processing cycles times
- Recyclable scrap
- Being less difficult to drill and machine

Product forms are still being developed and the most recently available prepregs are stiff and boardy. They lack the drape and tack needed for handleability (forms that handle well such as commingled fabrics will be discussed in Materal Forms in Section 2.5). Tack and drape in some thermoplastic prepregs are achieved by the presence of solvent, as is the case with some ployimides.

The "Wet" prepregs compared to "Dry" prepregs are described below:

(a) "Wet" Thermoplastic Prepreg Materials

- Wet matrices include KIII, AIX-159, Torlon 696, etc.
- Wet matrices have inherent tack and drape at room temperature
- Wet matrices may react chemically during processing (and in the past have been referred to as "pseudo-thermoplastics")
- Have half life and out time constraints
- Process like thermosets
- Have high volatile content, 12-25% by weight

- Some require post-cure for maximization of Tg
- Require volatile management during in processing

(b) "Dry" Thermoplastic Prepreg Materials

- More difficult to prepreg than "Wet"
- Dry matrices have no inherent tack and drape at room temperature
- Have no shelf life or time constraints
- Melt fusible, no chemical reaction
- Solution or hot melt impregnated
- Amorphous and/or semi-crystalline

One of the most critical factors is lack of an extensive database of performance properties over service time. Military aircraft structural applications are one of the major drivers to develop thermoplastics for use as high-temperature composite matrices and four major requirements are:

- High temperature capabilities (range 350°F (177°C)) under severe hot/wet environmental conditions
- Better damage tolerance in primary structures
- Easily automated in order to drive down manufacturing process costs
- Lower total part-acquisition and lifetime costs (including material, processing, and supportability)

To understand differences between thermoplastic matrices, an overview of properties dependent on their microstructure is shown in Fig. 2.2.4.

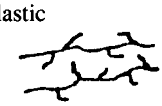
Morphology	Processing	Characteristics
Amorphous Thermoplastic 	Melt Fusion	Poor solvent resistance Lower temp capability Better formability
Pseudothermoplastic 	Condensation	Better solvent resistance Higher temp capability Some not reprocessible
Crystalline 	Melt fusion	Good solvent resistance Crystallinity dependent on processing
Liquid crystal 	Melt Fusion	Anisotropic properties Directionality of crystal dependent on processing Injection molding & extrusion

Fig. 2.2.4 Thermoplastics properties Depend on Microstructure

The characteristics of two major divisions (semi-crystalline and amorphous) of thermoplastic matrials are described below:

(a) Semi-crystalline matrices
- Have a definite melting point
- Better resistance to halogenated hydrocarbons and paint strippers
- Gradual loss of properties after Tg (glass transition temperature) is reached
- Density varies slightly depending on degree of crystallinity
- Mechanical properties may vary depending on degree of crystallinity
- Degree of crystallinity dependent on processing

(b) Amorphous Matrices
- No definite melt temperature
- Can be solvated for ease of fabric impregnation
- Free from the problems associated with crystallinity
- More susceptible to methyl chemical paint strippers

Fig. 2.2.5 compares the modulus vs. temperature curves for a typical amorphous polymer and a typical semi-crystalline polymer [note that in both cases the modulus has decreased by over an order of magnitude before the Tg (glass transition temperature) or the Tm (melting temperature) is reached]. Understanding the role of crystallinity in thermoplastics immensely improves the success of the engineer in both the design and manufacturing. Fig. 2.2.6 shows a number of available thermoplastic matrices.

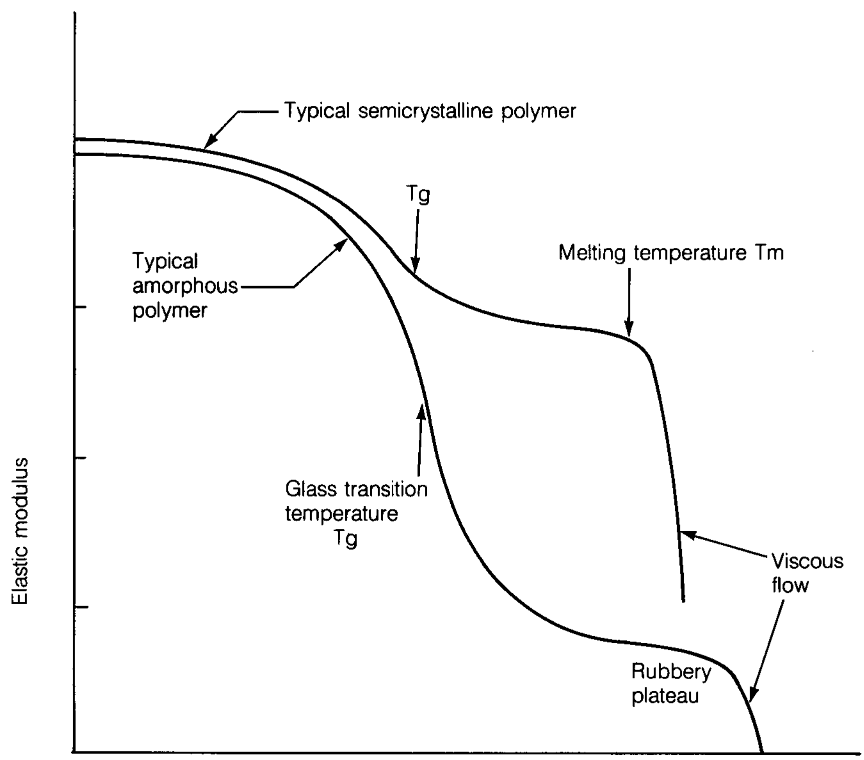


Fig. 2.2.5 Typical Modulus Vs. Temperature Curves for Thermoplastics

Polymer type		Serv temp	Material forms available	Remarks
Polyether- imide (PEI)	Amorphous	300°F	Fabric Film Molding compound Cowoven Commingled	• Good database • Unidirectional tape available, but poor quality • Low processing temperature • Good impact resistance
Polyether- sulfone		350 — 300°F	Unitape, Various fabrics Cowoven Commingled Film Molding compound Neat resin	• Very good processing characteristics • Solvent free prepreg
Polyarylene- sulfide (PAS)		300°F	Unitape Fabric Neat resin Molding compound	• Low processing temperature • Fiber/resin interface not optimal • Solvent free prepreg
Polyphenyl- enesulfide	Semi- crystalline	150 — 200°F	Unitape Fabric Discontinuous	• Very low processing temperature • Low service temperature
Polyether- etherketone (PEEK)		250°F	Unitape Fabric Commingled Film Molding compound Powder slurry Neat resin Towpreg	• Low moisture pickup • Large database • High processing temperatures • Solvent free prepreg • Processing window defined
Polyether- sulfone		350°F	Unitape powder slurry Film Fabric Molding compound	• Solvent free prepreg • Limited database
Polyimide	Amorphous >300°F		Unitape Fabric	• High service temperature • Long processing cycle • Has tack • Moisture sensitive • High volatile content • Limited reformability

Fig. 2.2.6 Thermoplastic Prepreg Matrices

Chapter 3.0

TOOLING

3.1 INTRODUCTION

In selection of tooling materials (see Fig. 3.1.1) one should be sensitive to the thermal expansion in the tool and try to match it to the coefficient of thermal expansion (CTE) of the composite. For elevated temperature forming and consolidation, where the tooling must go to the same temperature as the laminate, steel, carbon (graphite), or ceramic tooling material must be used. In forming operations where only the laminate is heated and then pressed into cold tooling, a variety of materials can be used, such as aluminum, wood and even rubber and silicone. Composite tooling differs from conventional (metallic) tooling as shown below:

- Tolerance build-up is much more critical
- The final machined dimensions of the tool are not necessarily the final dimensions of the composite part; the degree of disparity depends on:
  - Type of tooling
  - CTE characteristics
- Final part dimensions are those present at the ultimate gelation temperature of the matrix system

Tool Material	Caefficient Thermal Expansion	Heat Conductivity	Material Cost	Fabrication Cost	Durability
Aluminum	Poor	Good	Good	Fair	Fair
Steel	Good	Good	Good	Poor	Good
Graphite	Excellent	Good	Good	Good	Poor
Ceramics	Excellent	Poor	Good	Fair	Fair
Fiberglass Resin Composite	Poor to Good	Fair	Good	Good	Poor
Graphite Epoxy Composite	Excellent	Fair	High	Fair	Poor

Fig. 3.1.1 Tooling Material Guide

There are no hard and fast rules to ease the tooling selection decision, and while some guidance (see Fig. 3.1.2) can be offered, the most cost effective tooling choice is still evolving.

The following gives the rating of tooling properties (factor: 1 — lowest; 5 — highest):

TOOLING PROPERTIES	FACTOR
Dimensional accuracy	5
Dimensional stability	5
Durability	5
Thermal mass	4
Surface finish	3
Ease of reproducibility	3
Temperature uniformity	3
Material cost	3
Ease of tool fabrication	3
Ease of repair	3
Tool weight	3
Ease of inspection	2
Resistance to handling damage	2
Ease of thermocouple implantation	1
Release agent compatibility	1
Sealant compatibility	1

The following list shows the variety of possible tooling methods:

- Autoclave
- Out-of-autoclave
- Hard mandrel
- Washout mandrel
- Inflatable mandrel
- Pressure vessel
- Silicone rubber
- Press forming
- Diaphragm forming
- Mechanical pressure
- Integrally-heated tools
- Elastomeric tooling

Fig. 3.1.3 gives the pros and cons of each of the most commonly used tooling types.

It is obvious that tooliong for composites is a very wide field, involving many technologies. In only rare situations is there a well defined correct solution to a specific tooling problem. What is the best way to fabricate a component for one organization with considerable experience in a specific method, may not be optimum for another with a different background. Compromises are needed at almost every step of tool evolution depending on requirements, incentives, economic resources, schedules, and other such issues.

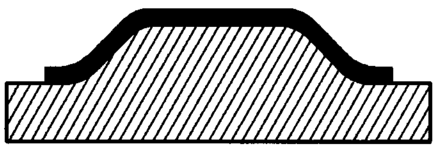
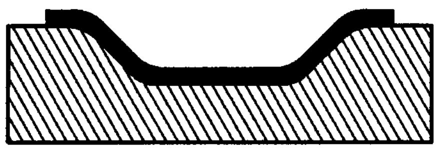
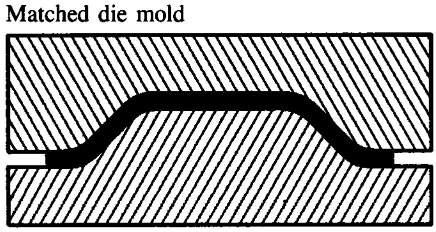
Type	Characteristics
(a) Male mold 	• Most commonly used for aircraft parts because of its low cost • Lowest layup cost • Small radius producibility ≥ .05 inch • Baseline (non-aerodynamic surfaces) • Surface control one side only • Localized control of vacuum bag surface
(b) Female mold 	• Limited use in contour applications because of bend radius • Highest layup cost • Radius producibility ≥ .25 inch • Localized control of vacuum bag surface • Surface control one side only
(c) Matched die mold 	• Used male/female tooling to control laminate thickness and is very expensive • Best thickness control • Highest tooling cost • Moderate layup cost • OML/IML control (smooth surface both sides)

Fig 3.1.2 Types of Tooling

Guidelines for composites tooling are generally the same as those for sheet metal forming dies or compression molding. Tool contact with the deforming material should occur in such a way that the sheet surface pressure is uniform at all times. In geometries where this is not possible, such as those where the loading direction is perpendicular to the surface, the use of flexible tool halves is recommended to provide a sort of hydrostatic pressure. Normally, tools should be designed with a draft angle of 1 to 2 degrees to counteract the effect of "closure" or "spring-in" after cure and to facilitate ease of part removal from the tools or dies.

The demands placed upon mold tooling for curing composite parts can be very severe; the ideal tooling characteristics for composites are given below:

- CTE characteristics compatible with part to be produced
- Able to withstand severe temperature and pressure conditions without deterioration
- Dimensional stability
- Low cost
- Durable
- Reproduce pattern with high dimensional accuracy
- Retain mechanical properties at high temperatures

Type	Potential uses	Pro	Con
Autoclave	• Large components • Low volume production • Honeycomb sandwich assemblies • Cocured parts • Parts having vertical walls • Bonding	• Low cost • Internal heating possible • Undercut feasible • Vertical walls attainable • Versatility, particularly with large components • Complex, cocured parts attainable • Thermal expansion can be made to match part	• Low production rates • High labor cost due to ancillary material layup • Loose dimensional control of bag surface • Low molding pressure, relative to matched dies require more generous radii • Curing temperatures limited by ancillary materials unless interally heated tools are used with insulation installed between bag and layup • More process variables involved than with matched dies • Bag failure usually causes part to be scrapped
Matched metal dies	• Relatively small parts • Both surfaces dimensionally controlled	• High productivity • Good dimensional control • High molding temperatures • Good quality surfaces on all faces • High fabrication pressures • Durable • Internal heating feasible • Good thermal response and control • Compression molding tool technology available • Minimizing ancillary material use	• High cost due to machining, stops guides etc. • Tool thermal expansion different from composite • Limited ability to selectively reinforce • Undercuts require multi part tools • Draft angles required where vertical wall preclude part removal from tool • Large components present tool flexibility, heating uniformity and air-volatile removal difficulties • Difficult to repair or modify
Elastomeric	• Allows complex geometries • Large components feasible	• Considerable part design flexibility • More complex parts feasible than with matched metal dies due to casting of elastomeric elements • Ability to layup on numerous elastomeric mandrels and install these in the metal tools allows complex, parts to be made	• Limited life • Volatile and air removal less than ideal • 500°F processing limit • Low conductivity of elastomeric elements can cause undesirable thermal gradients in the part
Monolithic graphite	• Tight dimensional control of complex components • Rapid cure cycles (high heat up rates) • Prototype parts	• Low CTE, matched to graphite fiber composites • Temperature capability 600°F • Lower cost than metals • Easily machined in specialized facility • High thermal conductivity • Easy part release • Easy to repair or modify	• Susceptibility to impact damage • Special precautions needed when machining • Not suited for matched die molding
Ceramics	• Tight dimensional control of high temperature components	• Can be cast into complex shape • Low CTE which can be controlled • Electric and fluid heating systems easily cast into tool • Temperature capability 600°F	• Susceptible to impact damage • Difficult to repair

Fig. 3.1.3 Summary of Tooling Types (pros and cons)

Concurrent development of composite design and the tooling used to build it is the soundest foundation for product success. When the complexity of a part indicates serious problems, design modification should be sought early in the development phase. This allows mutually acceptable compromises between the design and manufacturing, to the long term advantage of both. However, after production tools have been made is almost always too late (due to both schedule and cost) to make major changes to the design. Much basic information on the design of tools for metallic structures is relevant when designing tools used to fabricate composites.

The method of heating the tools and ensuring their satisfactory thermal performance also strongly influences tool design. Inability to provide the desired temperature time history throughout the cure cycle is a basic fault of some tools. In most applications, each tool is used in conjunction with a number of ancillary materials (as discussed in Chapter 4.0) such as breathers, bleeders, bags, and sealants; their part must be considered in tool design.

Tooling cost should be kept as low as possible, particularly in prototype and research or development programs. The use of sculptured metal shapes should be limited. Composite tooling can also be expensive if tooling cast form molds is used for lay-up and cure; the final tool has to be high-temperature resistant and compatible in thermal expansion.

Life cycle costs are important to consider in tool design and tooling should be a non-recurring cost in the life of a fabrication project. Short life cycle tools or tools requiring a high degree of maintenance result in recurring costs which burden the total product cost.

Tool repair procedures must be sufficient to maintain:

- Strength
- Heat transfer
- Dimensions of the original tool
- Surface finish characteristics of the cured part

Nevertheless, tool design, construction, and operation must be consistent with the component being produced and the characteristics of the composite materials being used and their related fabrication and cure operations. It should be kept in mind that tools for composites are unique and careful consideration should be given to tool design during the fabrication of any composite component.

A tooling design should result from team effort and communication between the design, tooling, and manufacturing disciplines will minimize problems and provide a superior end product.

The following three aspects are integral components of composite structures:

- Design
- Tooling
- Manufacturing

These disciplines must work together throughout the development phase of any product, from idea inception to its culmination as hardware. The primary areas in which inter disciplinary agreement is needed concern:

- The trade-off between simplicity of tool design and number of parts produced
- The ability of composite materials to provide part count reduction

The overall reduction of part count can have an accompanying cost reduction benefit for the final structure through increased ease of assembly and minimized time flow in production. Full communication between disciplines provides optimum conditions to achieve this trade off.

All tooling systems discussed in this Chapter ralate very closely to the manufacturing methods dicussed in Chapter 4.0.

Tooling Materials

In complex, highly curved components a serious degradation in composite strength and dimensional accuracy can result when there is a serious mismatch in tool and composite CTE. Therefore, for parts that must meet close dimensional control or part mating requirements, the choice of the tooling material becomes very important and thermal change must be calculated into all tool dimensions. A prime consideration for selection of materials for fabrication of large tools used for curing composite structures is compatibility of thermal expansion between the tool and part.

Tooling materials currently used are

- Aluminum or steel — fabricated by standard metalworking techniques
- Electroformed nickel
- Composite mold tooling — Laid up and cured over a master model to produce a mold tool

Fig. 3.1.4 gives both the coefficients of thermal expansion and coefficients of thermal conductivity for most commonly used tooling materials. To compare with materials, see Fig. 2.1.3 of Chapter 2.

Material	Thermal Conductivity (BTU-in/ft^2-hr-°F)	Coefficient of Thermal Expansion (Micro-in/in/°F)
Graphite	400	1.5-2.0
Aluminum	1395	13.0
Steel	350	6.7
Nickel	500	6.6
Carbon-Fiber/Epoxy	24-42	0-1.5
Fiberglass/Epoxy	22-30	7-13
Ceramics	10-80	3-6
(MgO, Al_2O_3, Gypsum)		

Material	Apparent Density (g/cm^3)	Specific Heat (Cal/Gr-°C)	Thermal Mass (Cal/cm^3-°C)
Graphite	1.78	0.25	0.44
Aluminum	2.70	0.23	0.62
Steel	7.86	0.11	0.86
Nickel	8.90	0.11	0.98
Carbon-Fiber/Epoxy	1.5	0.25	0.38
Fiberglass/Epoxy	1.9	0.3	0.57
Ceramics	1.6-3.9	0.84-1.50	1.2-5.3
(MgO,Al_2O_3 Gypsum)			

(Source: Stackpole Carbon Co.)

Fig. 3.1.4 Typical Properties of Tooling Materials

The basic material characteristics relevant to tooling for composites depends to a great degree upon each application; factors determining choice of tooling materials are:

- Thermal Expansion compatibility
- Thermal Conductivity
- Accuracy required
- Strength
- Ease of tool fabrication
- Shop capability
- Cost per part
- Durability

Material	PRO	CON
Aluminum	• Machineability • Thermal conductivity • Pressure from expansion • Low weight and mass	• Dimensional Stability • Becomes soft when heated over 350°F • Easily distorted • Incompatible coefficient of thermal expansion
Steel	• Durability • Surface finish	• Warpage • Thermal expansion • Weight
Monolithic graphite	• Low expansion • Dimensional stability • Easily machined • Heat resistant • Thermal conductivity • Cost (for prototype parts)	• Porous • Soft surface • Bonding • Low strength • Needs back-up structure
Ceramic	• Low expansion • Dimensional stability • Heat resistant • Low shrink casting	• Porous • Soft surface • Machineability • Low strength • Thermal conductivity
Silicone rubber	• Pressure from thermal expansion • Out of autoclave cure • Can be molded to any shape • Low cost	• Hard to control, predict and measure pressure • Loses dimensional stability with repeated use

Fig. 3.1.5 Tooling Material Selection

Material	PRO	CON
Graphite/ epoxy	• Excellent dimensional stability Stability • Good heat-up rate • Lightweight • Very compatible coefficient of thermal expansion • Low denisty • Ease of construction (plaster model) • Low cost tooling	• Durability • Limited strength at high temperatures • Must build master model • Not feasible for molding cocured stiffened panels
Electro-formed nickel	• Very smooth and scratch-resistant mold surface • Coefficient of thermal expansion (CTE) about 40% that of aluminum • Rapid heat-up/cool down • Light weight • Good repairability • Good release properites • Low cost to duplicate tools	• Long lead-time • High fabrication cost • No long-term durability • Substructure very heavy on large tools
Invar	• Low CTE • High thermal conductivity • Durable	• High material cost • Long lead-time • High fabrication cost
Avamid-N	• Good CTE match • Low fabrication cost	• Limited strength at high temperature • Limited high temperature capacility

Fig. 3.1.5 Tooling Material Selection (cont'd)

- Repairability
- Life Assessment
- Tool mass

High thermal conductivity is desirable:

- To minimize large thermal gradients in the tool and the part being fabricated
- When high temperature thermosets (i.e. polyimides) and thermoplastics are being cured, rapid rise in temperature may be required

Fig. 3.1.5 gives a tooling material selection comparison.

Tool Fabrication

The conventional sequence of composite tool fabrication:

- Master model (see later discussion in this chapter) is typically fabricated from plaster which computers can be used directly to machine the master model by numerical control (NC) process
- Plastic faced plaster mold is made off master model
- Graphite epoxy is laid up on plastic faced plaster mold to produce working tool

Alternative tool fabrication procedures:

- Machining of tool directly from metal
- Machining of tool directly from solid monolithic graphite
- Fabrication of tool directly from metal parts placed on a graphite base
- Casting of tool, e.g., ceramic mixture material, directly from master model

The following are general requirements for a tool design:

- Tool should extend a minimum of 2 to 3 inches (5 to 8 cm) beyond the edge of the part (actual edge designated by engineering drawing)
- Provision should be made for vacuum attachment
- Edge sealing provision should be considered to minimize the use of tape sealants (bagging problem area)

Tools always need to be adequately supported; metals used in tooling are chosen because they are inherently strong and low cost. The tool can be strengthened by a conventional "egg crating" constructed backup as shown in Fig. 3.1.6 which has two advantages:

- Lighter total weight
- Rapid heat to the back side of the tooling

For non-metallic tooling, however, the "egg-crating" or similar backup structures should be manufactured from non-metallic materials as a separate structure from the tool. It is generally more convenient to produce a return flange at right angles to the face of the tool, thereby strengthening the tool and providing support stand.

The backup structures added to provide stiffness and strength must be designed to avoid thermal distortions of the tool and localized temperature control problems.

Heating Systems of Tools

It is clear that tool thermal response must be accurately determined since temperature control is so important in the curing of composites. To accomplish this the selection of the heating systems and the tools response to it must be carefully evaluated. The three basic heating systems are:

(1) External heating by hot gases, typified by the conventional operating mode of autoclave or hydroclave (by heated water, oil, etc.) and these methods involve quite sophisticated calculations that depend on:
   - The operating characteristics the autoclave heating fluid system
   - Tool size
   - Material and part shape

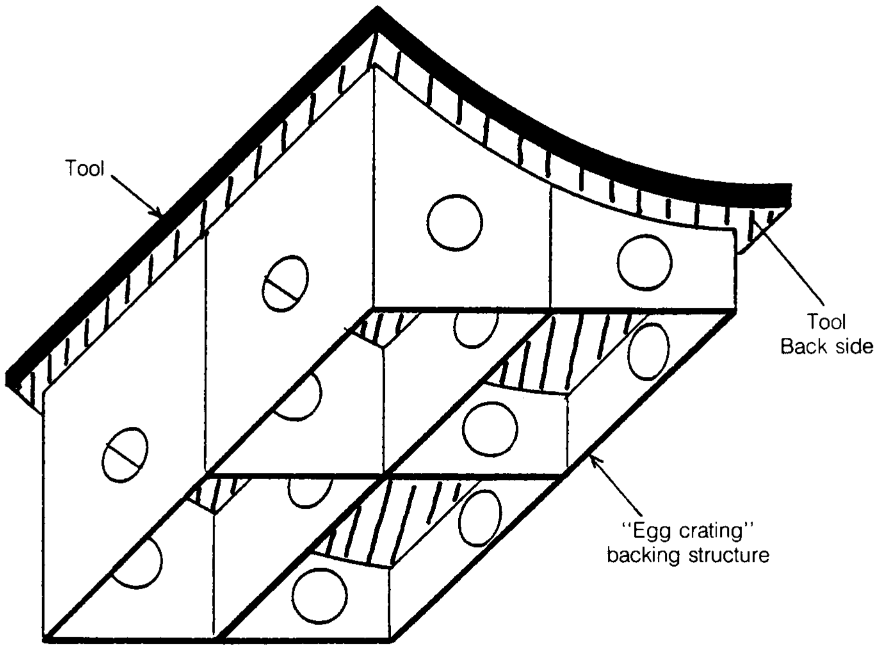


Fig. 3.1.6 Underneath View of a Tool to Illustrate "Egg Crating". Multiple Holes Allow Gas Flow to Rear of the Mold Surface.

- Heat capacity of all the tools in the autoclave during the run
- Position of the tools in the hot gas stream

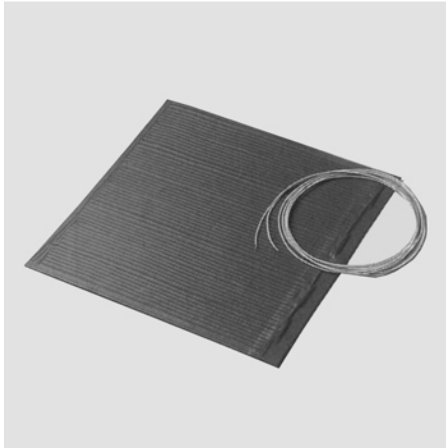
A most important facet of this is developing a thorough understanding of the thermal and flow rate characteristics of each autoclave.

(2) Electrical heating of either the tool itself or the platen in intimate contact with the tool which can be accomplished by:

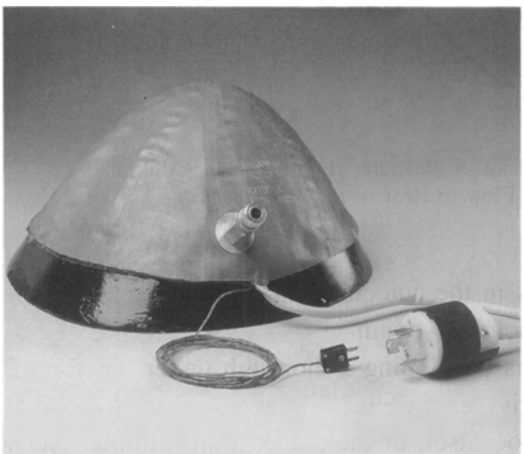
- Computing tool volume, weight and surface area
- Calculating the heat required
- Determining the heat lost from the un-insulated side of the tool
- Adding together the factors mentioned above provides the total number of BTU required to heat the tool to the desired temperature

This method also involves custom-designed heating elements and blankets for curing composites. The heating systems are available in single or multiple zone units, with temperature capability to 1400°F (760°C). Instead of heating the whole environment, a blanket can be used to heat only the part, saving energy:

- Silicone heat blanket, shown in Fig. 3.1.7(a), provides uniform heating up to 450 — 600°F (230 — 320°C)
- Insulating blanket, shown in Fig. 3.1.7(b), can be used with heating blanket to reduce heat lost to the environment and it can withstand up to 900 — 1400°F (480 — 760°C)
- Fig. 3.1.7(c) shows a reusable vacuum bag and heating blanket in one, which heats up to 450°F (230°C)



(a) Heating blanket up to 450°F



(b) Insulating blanket up to 1400°F

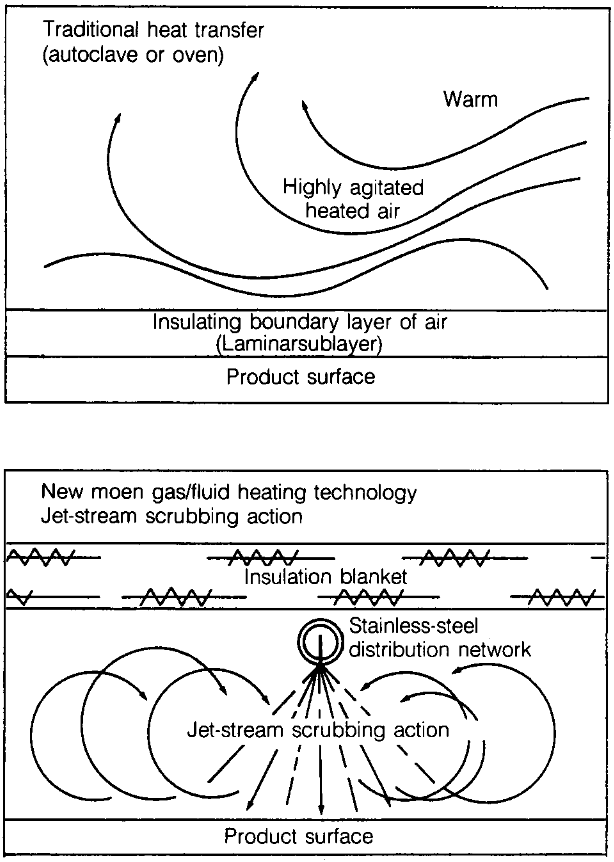
(c) Experimental compled curvature vacuum blanket up to 450°F

By courtesy of Briskheat Corp.

Fig. 3.1.7 Heating Blankets

(3) Fluid heating of either the tool or platen interior:
Steam or heated oil are the most commonly used media which are circulated within coils buried in the tool. Essentially the heating is controlled by the length of heating coil per unit volume of tool. The oil heating devices, sometimes known as Hydrotherm units, are programmable to provide heating or cooling between the temperature range of roughly 60°F (15°C) to 375°F (190°C)

(4) High-velocity jet stream heating (Moen system):
With the Moen system, shown in Fig. 3.1.8, the laminate part is subjected to high-velocity jets of air and the heat source is brought to the part, rather than the part being brought to the heat source; this heats the part faster and more efficiently and thus, requires less energy.



By courtesy of Heat Transfer Technologies

Fig. 3.1.8 Moen System-High-Velocity Jet-Stream Heating systems

This heating system uses a series of thin-walled tubes to distribute hot or cold jets of air through a series tiny holes along the tubes, and the tubing, serpentined over the surface of the tool, acts as a framework for the insulation barrier that surrounds the framework of tubes. This system can provide a temperature range between 300 to 2000°F (150 to 1100°C).

(5) Microwave and induction heating systems have been used successfully. These methods have been used on small parts and tend to be geometry dependent but have found little favor in the large structure field.

All of the above systems except item (5) accomplish the polymerizing step by heating the part from the outside in. This is a very inefficient method for thicker laminates and can result in laminate structures which are non-uniform. This is because the outside layers gel before the entrapped air and excess matrix at the interior of the laminate can be removed.

The heating method of item (5) cures the laminates from the inside out by activating molecules creating internal heat resulting in the polymerization of the matrix.

3.2 TOOL DESIGN CONSIDERATIONS

The three most significant factors which control final tool design concept selection are

- Cost
- Tool service life
- Dimensional stability

This Chapter will discuss some of the principal factors which should be considered during the planning and design stage for tooling to be used in the fabrication of composite structures.

(1) Thermal Expansion

The increasing use of woven and unidirectional tape, which have widely varying CTEs, has emphasized the necessity for thermal match in tool design. The problems encountered are most acute in the fabrication of long relatively slender composite structures and also complexly shaped components.

Steels and aluminum have a CTE around an order of magnitude greater than most carbon/graphite composites. This means that the metal tool contraction during cool down from the peak cure cycle temperature can induce severe residual or "built-in" strains in the component. If this reduce the structural capability it may be necessary to use low CTE tooling such as carbon/graphite composite, monolithic graphite, ceramic, etc. Unacceptable dimensional tolerances can also arise because of a CTE mismatch between the composite and its tooling. Therefore, low CTE materials for composite tooling may be required to resolve this problem.

(2) Part Size and Configuration

Occasions may well arise when the component to be fabricated is too large to fit in the available processing equipment, be it

- Autoclave
- Press
- Oven

In this case, there is little choice but to utilize a self-contained tool (the tooling and processing equipment for a specific component is built as a single unit, as discussed later in this Chapter) to fabricate the that part. The self-contained tool is facility independent. There is also a size limitation on the use of matched metal molds. Large matched metal tools are heavy, have excessive heat capacity or may be too flexible for press applications.

Tool size or complexity may require a redesign to create smaller, simpler parts which more joints and thus greater assembly costs. The compromise of design and tooling properties must be made to achieve subsystems goals varies from case to case.

Part configuration will affect the tooling selection because of the varying amounts of pressure needed to produce:

- A complexly shaped skin panel
- A thickness variation
- Taper or ply drop-offs

(3) Part Tolerance

In properly designed composite structures, the requirements for close control of part tolerances is held to a minimum. Composite aircraft components are usually tooled to the surface having an appearance or aerodynamic smoothness requirement. There are cases where components are tooled to the faying or mating surface to be subsequently mechanically fastened or adhesively bonded. For example, if only one face of the component is a controlled dimension, it is chosen as the tool face. The other surface dimension is not controlled precisely. However, there are designs where it is necessary to closely control both face dimensions. This can only be done to close tolerance by using a matched mold (see Fig. 3.1.2(c)) of some type.

Matched die or press to stop tooling will, of course, result in part thickness equivalent to the tool cavity dimensions; such tooling must be capable of compensating for the material bulk factor (thickness of preform lay-up/thickness of cured component). The bulk factor be as high as 125% even with intermediate debulking cycles during the lay-up.

On a part or assembly the tolerances play a very important role in the type of tool selected and the cost of the tool. Practically, the engineering tolerance used for a cured laminate is $\pm 10\%$ of the part thickness. Choice of tooling must anticipate assembly requirements. To summarize:

- Part tolerance is an important factor in determining cost and the type of tool selected
- Close tooling tolerances when producing thin laminates, e.g., 0.025 inch (0.63 mm), is expensive and can be impractical
- Tolerance accumulation affects composite assemblies to a greater extent than sheet metal assemblies

(4) Tool Life Expectancy

It is universally held that long-run production tooling must be made of steel. Aluminum tooling made of heavy roll formed and machined plates is the second choice. Non-metallic tooling is generally not as desirable, but is acceptable to development programs in which few units are to be fabricated.

(5) Surface Finish

Polished tool surfaces (RM63 or less) are generally not required for composite structures. Almost invariably, the preferred tool surface a parting film such as Teflon, fluorinated elastomer, or Teflon impregnated fabric. In addition, the use of peel plies, a sacrificial piece of fabric such as nylon cloth, is becoming more prevalent. This material acts as a surface bleeder in net cured resin systems and is removed just before bonding or painting.

Tooling surface requirements occur when:

- Tooling surface location is an important factor in final assembly fit and function
- Specification of surface smoothness in addition to tooling surface may be necessary to meet aerodynamic requirements on some designs such as high performance wing surfaces

(6) Repairs

All types of tooling used for composite structure curing can be repaired

- Steel and aluminum tools are repaired by welding and grinding to restore contour. Minor surface damage in steel or aluminum tools can also be repaired in using fill-in resins
- Composite material tooling can be repaired by grinding out the damaged area to ensure that a clean solid surface is reached and then rebuilding the area with the same fabrics and resin

(7) Process Effects

Tool design, construction, and operation must be consistent with the number of component to be produced and the characteristics of the structural materials being used and their related fabrication and cure operations.

The following must be taken into considerations:

- If the tooling is to be used in a blanket press or autoclave, the tool surface, pressure or vacuum bag, and bag seal must be vacuum tight. This characteristic must be designed and built into the tooling from the beginning of the tool design, since it is virtually impossible to effect a permanent leak repair.
- If internal component pressures are to be supplied by means of solid silicone elastomeric blocks, direct heating of the blocks by means of cartridge inserts may be required
- If matched molding or press operations are used, adequate means of press mounting, tool alignment, and means of heating and cooling must be incorporated

Processing considerations include:

- Temperature:

— 350°F (180°C) — must thermoset materials

— 350 to 700°F (180 to 370°C) — thermoplastics or polyimides

— Above 700°F (370°C) — thermoplastics, polyimides, etc.

- High temperature tooling [i.e., above 350°F (180°C) for curing or consolidating materials of thermoplastics, polyimides, etc.]:
  - — Expensive
  - — Requires heat stable tooling materials
  - — Thermal effects must be reduced
  - — See Fig. 3.2.1 for assessment of tooling materials suitable for high temperature composites
- Pressure:
  - — Vacuum only
  - — Pressure chamber for higher pressure than that produced by vacuum
  - — Press
  - — Rubber mandrel thermal expansion
  - — Aluminum block (or modules) thermal expansion

Tooling material	Process compatibility	PRO	CON
Steel	• Autoclave • Press	• Good thermal conductivity • Durable	• Warpage at high temps • High CTE • High fabrication cost • High denisty
Invar	• Autoclave • Press • Diaphragm	• Low CTE • High thermal conductivity • Durbale	• High material cost • Long lead-time material • High fabrication cost
Titanium	• Autoclave • Diaphragm	• Low CTE (closely matches composites) • Good thermal conductivity • Durable	• High material cost • High fabrication cost • Limited experience as a tooling material
Ceramic	• Autoclave • Diaphragm	• Low cofficient of thermal expansion • Low cost material • Low fabrication cost	• Low thermal conductivity • Fragile • Low fabrication cost • High density
Monolithic graphite	• Autoclave • Diaphragm	• Low CTE • Good thermal conductivity • Low density	• Fragile • Limited vacuum integrity • Special machine handling equipment required • Moderate fabrication cost
Aluminum	• Diaphragm • Press	• Low cost material • High thermal conductivity • Medium density	• High CTE • Limited strength at high temperatures
Avamid-N	• Autoclave	• Good CTE match • Low fabrication cost	• Limited strength at high temperatures • Limited high temperature capaibility

Fig. 3.2.1 Assessment of Tooling Materials Suitable for High Temperature Composite

(8) Tool Proofing
Before a tool is used for the fabrication of parts, a composite thermal survey and contour check should be performed. This can be accomplished by fabricating a representative part to the approved process specification.

(9) Tool handle provisions
If a tool weighs over 40 lbs, handling features are required which include lifting rings, fork-lift handling features and castor assembly. However, even the lightest weight tools made from composite material require special protective handling, because they are more prone to handling damage than their metal counterparts.

3.3 METALLIC TOOLING

Technological advances have made it possible for some metals to be competitive in composite applications. Metal has always offered advantages of durability and, under good maintenance, a virtually unlimited tool life. Several metallic allows are also free of the CTE mismatch that plagues conventional steel and aluminum tooling.

High CTE tools may cause micro cracking in the composite part or fiber distortion in high temperature part. This happens during the time span right after composite gelation occurs. In many design cases thermal expansion of metal tools is used for compaction of laminates and the thermal contraction of the tool aids in the removal of the part from the tool.

Metal tooling has proven to be the most popular and heat transfer is another important characteristic which makes metals very effective. However, as composite structures grow in size and complexity, the differences in CTE make it difficult to control the final dimensions. Metal tools also have high thermal inertia which means that heat-up/cool-down takes time and energy.

However, when judged strictly on the basis of durability in mass production, metallic tooling often turn out to be the most economical choice.

Aluminum

Aluminum tooling is one of the most widely used tooling materials in composite manufacturing and it is generally used for flat laminates or small and slightly contoured laminates. It is readily cast and machined, light in weight to facilitate handling, quite cost effective, and has excellent thermal conductivity for heat transmission during forming and curing. For many applications aluminum is the material of choice for curing composites at temperatures below 400°F (200°C).

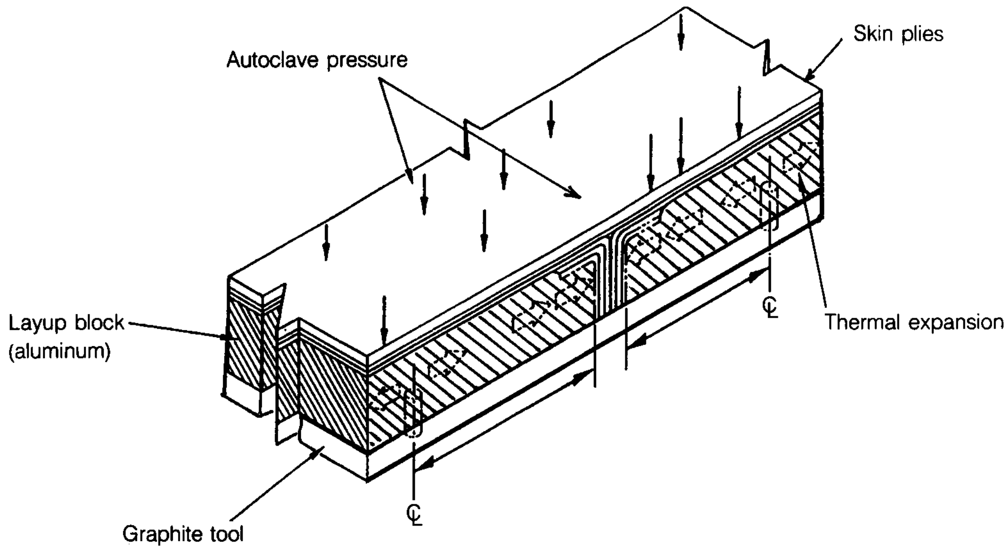
Aluminum material tools are satisfactory for simple parts without tight tolerances. Since aluminum has a relatively high CTE, this can be a problem or an advantage, depending upon the application, and this factor must be accounted for. It can be used as "modules" to create compaction pressure to fabricate skin stiffener flanges during an elevated temperature cure (see Fig. 3.3.1) a method similar to that of elastomeric rubber tooling except using aluminum. The high CTE is not a problem for the fabrication of almost planar structures such as wing cove panels.

Advantages of aluminum tooling are:

- Low cost
- Easily machined to complex shapes
- Excellent thermal conductivity for heat transmission during forming and curing
- Lower weight compared to steel
- Easily repaired by welding and grinding to restore contour

Aluminum is:

- Not applicable for highly compound curved parts due to its relative high CTE
- Not suitable for curing above 350°/400°F (177°/204°C) or consolidation of such materials as thermoplastic composites



[Stiffeners and skin are compacted from thermal expansion aluminum blocks (or modules)]

Fig. 3.3.1 Aluminum Blocks Create Compaction Pressure by Their High CTE

Steel

Steel is a proven tooling composite material since large numbers of composite parts, both structural shapes and skins, have been made on steel tools. Steel has been used as the predominant tooling material because of availability, low cost, and a more compatible CTE compared to aluminum. Stainless steel is used extensively when severe radius forming is required. For a large tool, steel (or titanium) is a better choice than aluminum although still not totally compatible with the thermal expansion of composite structures.

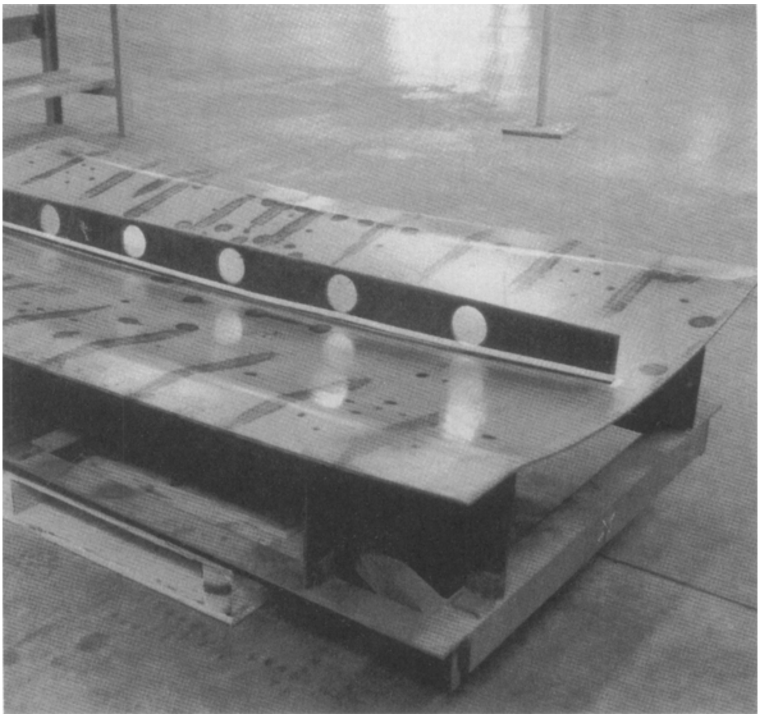
Steel has following advantages:

- High Durability
- Weldability
- Repairability
- Vacuum integrity

Concerns are:

- High initial cost
- Difficulty of forming into complex curves and shapes
- Weight

In practice, to minimize the mass of the steel tool for relatively simple tools, sheet or plate shock can be machined to the desired configuration. If this is not feasible, rolled, or formed and/or welded and machined sheet stock as shown in Fig. 3.3.2, can be used to build up the mold surface (machined stock may be needed if close tolerances are required) tool structure. On large tools, backup structure of heavy steel tubings and angles are frequently required where these are attached to the mold surface. It is recommended to use an insulator between the mold surface and the backup structures to prevent the heat sink capacity of the backup structure from disturbing the thermal gradient over the mold surface.



By courtesy of The Boeing Co.

Fig. 3.3.2 Rolled and Formed Steel Sheet Stock Tool for Consolidation of Thermoplastic Composites

Invar

Perhaps the best known of the metal tooling materials is Invar 36 (36% nickel) or Invar 42 (42% nickel) low-carbon austenitic steel alloy with very low CTE (from 0.5 – 6×10^{-6}in/in/°F). Invar refers to the metal's "invariable" dimensional properties. Invar is durable as steel. It has, in fact, all of steel's advantages, plus the added bonus of a CTE which matches composite materials.

(1) Invar 36 boasts a CTE of 1.5×10^{-6}in/in/°F, more suitable for thermosetting composite materials.

The material properties of Invar 36 as follows:

Density lb/in^3 (g/cc)	0.29 (8.12)
Thermal conductivity (32 to 212°F) BTU/ft^2/hr/°F/in	73
CTE (10^{-6}in/in/°F):	
0.5 (room temperature to 200°F)	
1.5 (200 – 400°F)	
2.7 (400 – 600°F)	
6.0 (600 – 800°F)	
Tensile strength (ksi)	70
Yield strength, (ksi)	40
Elongation	4%

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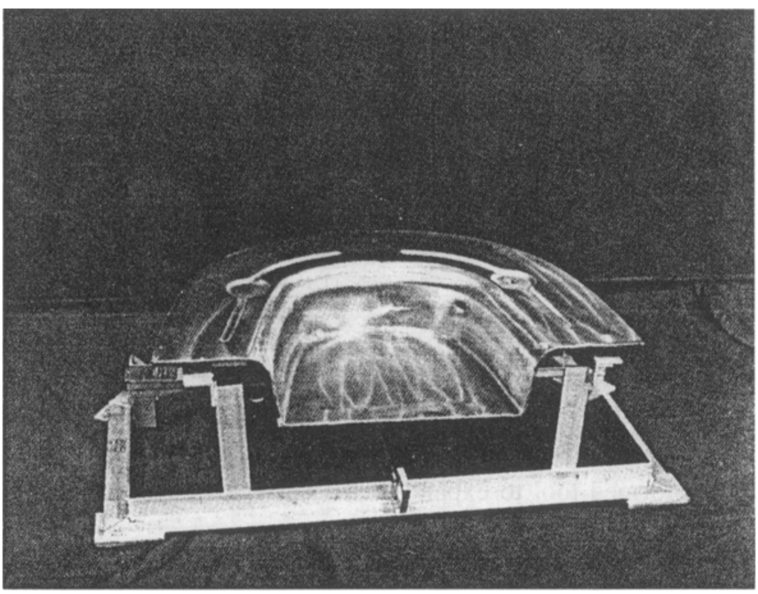
(2) Invar 42, with CTE of 3.5×10^{-6} in/in/°F, can be used to build tooling for high temperature thermoplastic composite materials in the 500 to 800° (260 to 430°C) range.

Considerations in its use are:
- High initial cost
- Low thermal conductivity
- Weight
- Welding has been troublesome.
- Vacuum fittings (may have to be custom made from Invar in order to avoid leaks at the fittings)

Electro-deposited Nickel

Electro-deposited nickel tool (or electroformed or electroplated nickel tooling) is done by electro-depositing a platable metal over a mandrel or plastic model that is subsequently removed. Process steps for producing an electro-deposited nickel tool are shown in Fig. 3.3.3.

Advantages include:
- Durability
- Good part release, damage resistance, and vacuum-leak resistance
- Relative ease of repair via soldering or welding
- Tooling to be fabricated is limited in size only by the size of the electroforming tanks



(a) Electroformed nickel mold for autoclave production of composite parts.

Fig. 3.3.3 Electro-Deposited (Electroformed) Nickel Tooling

Chapter 4.0

MANUFACTURING

4.1 INTRODUCTION

Manufacturing processes and tooling are the elements which control the success and cost of a composite component and it is therefore mandatory that they should be considered as an integral part of the design process. This Chapter, which concerns composite manufacturing processes, will concentrate on organic matrix materials because they have been widely used as airframe primary structures for several decades. The cure (thermoset)/consolidation (thermoplastic) phase of the composite manufacturing process, as shown in Fig. 4.1.1, involves application of heat and pressure to a layup laminate in controlled cycle. Manufacturing of other composites, such as metal matrix, carbon/carbon, and ceramic matrix composites, which have limited applications and were briefly discussed in Chapter 2.0 will not be addressed in this Chapter.

In this chapter, only those manufacturing methods which are applicable to airframe structures will be discussed.

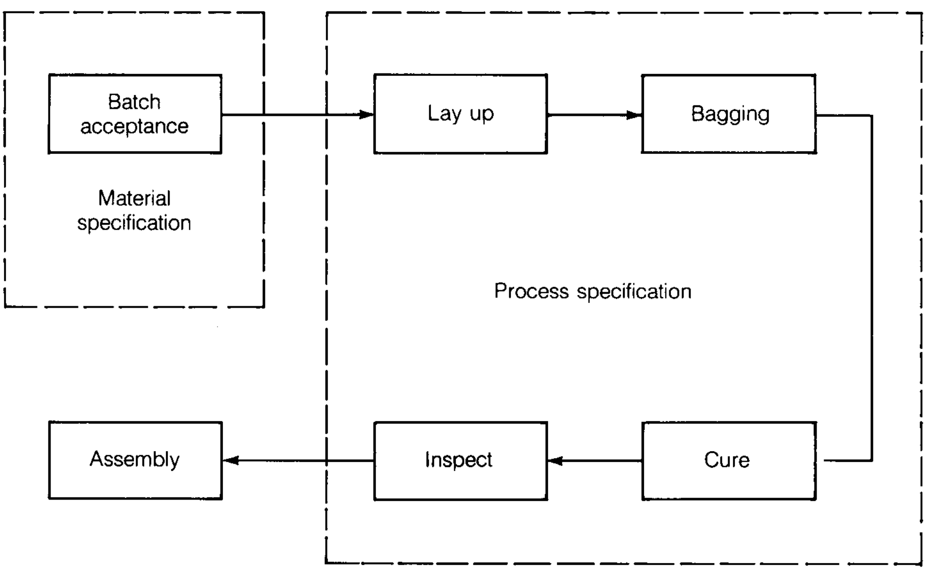


Fig. 4.1.1 A Flow Chart of Composite Laminate Part Fabrication

In composite manufacturing, it is difficult to draw a clear line between manufacturing and tooling systems because they are so closely related each other. Therefore, the tooling information in Chapter 3 should be considered simultaneously with composite manufacturing. The basic problem facing manufacturing engineers is producing composite hardware at an acceptable cost while ensuring its reliability. As matter of fact, almost any composite hardware can be fabricated if economic constrains are removed.

Preparation for a composite manufacturing program should include the early involvement of manufacturing personnel including:

- Active participation in the development of early design concepts — review and evaluation of preliminary engineering drawings
- Support of engineering in selection of materials:
  - Fabrication of test panels
  - Fabrication of basic part configuration specimens
- Coordination of manufacturing and quality assurance organizations;
  - Formulation of tooling concepts
  - Development of inspection, property tests, and non-destructive inspection (NDI) criteria
  - Establishment of the level of planning documentation required
  - Issuring cost and schedule quotes
- Development of a manufacturing plan
- Fabrication of developmental and engineering test configurations
  - Process development specifications
  - Engineering test requirements
- Proof of production tooling concepts
  - Production readiness verification tests
  - Full scale tool prove acceptance
- Support of production during fabrication of hardware

Usually, composite components contain fewer parts than their metallic counterparts. This feature, plus the reduced number of mechanical fasteners required in most composite designs, is a basis for cost reduction. Of course, the part size, geometry, complexity, and required quantity are all considerations in the selection of a fabrication process.

Generally, the most labor-intensive step in composite fabrication, as shown in Fig. 4.1.2, has been the lay up of uncured tape plies, which is normally done by hand. On parts amenable to the tape form, advantage can be taken of the automated methods of dispensing and laying tape with pre-determined orientation and ply sequencing. Some companies have directed a large portion of their research efforts into the manufacturing process so that the implementation of automated systems could be cost-effective. Some of these advances are beginning to be incorporated into the industry's building blocks for the so-called "Factories of the future". One of the processes which has received a great deal of attention in manufacturing research is automated composite systems. The automated system is a must in composite manufacturing to reduce cost.

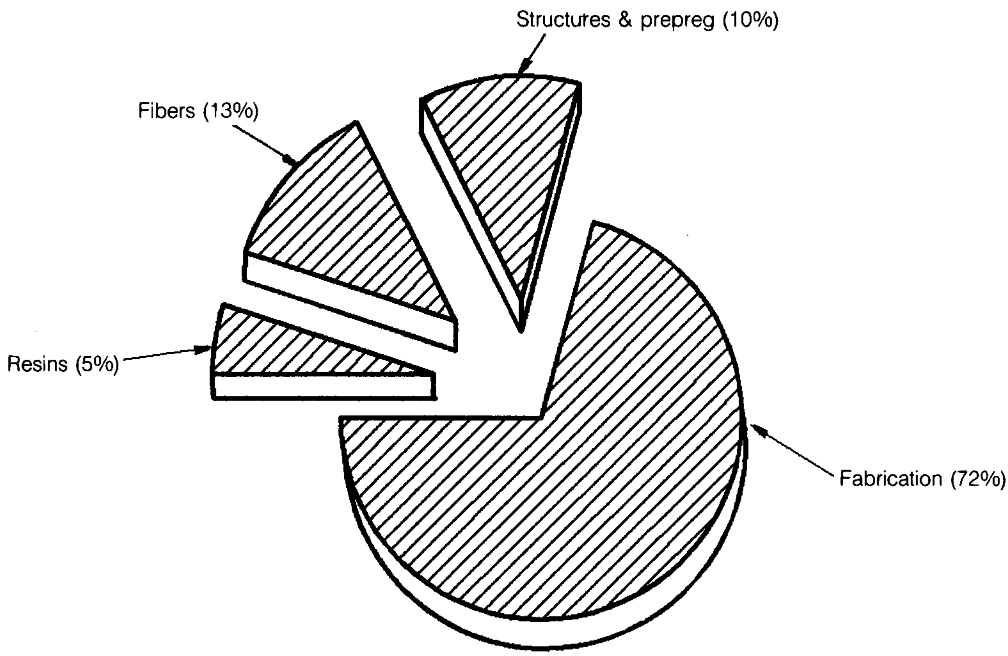


Fig. 4.1.2 Today's composite Fabrication cost Distribution

Material Selection

When selecting composite materials for manufacturing, the following items should be considered:

- Cost
- Ease of processing, fabrication, and handling
- Immediate or near term commercial availability (except for developmental programs)
- Multiple material sources
- Potential to be used in an automated manufacturing process

Material specifications generally include:

- Qualification or source approval requirements
- Incoming batch acceptance requirements

The selection of material forms is quite varied:

(a) Material forms:
- Matrix types
- Reinforcements (fibers):
  - — Individual fibers
  - — Chopped fibers
  - — Woven fabric
  - — Tape
  - — Mat
  - — Preforms: 2-D, 3-D, multi-directional preforms
  - — Others: knits, braids, hybrids

(b) Wet layups
(c) Prepregs:
- Solvent coated
- Hot melt coated
- Powder impregnated (thermoplastics)

(d) Commingled (thermoplastics)
(e) Fiber bundles or shapes

Fabrication methods for thermoset and thermoplastic composites may appear to be similar. Heat and pressure are applied to both materials to transform basic prepregs and other material forms into cured or consolidated laminates. However, there are some major processing differences as summarized below:

PROCESS	THERMOSETS	THERMOPLASTICS
Chemical reaction	Yes	No
Temperature range	250-400°F (121-204°C)	500-800°F (260-427°C)
Cycle time	3-7 hours	Can be less than 30 minutes
Viscosity	Low	High
Pressure required	50-100 psi	200 psi and higher
Fabrication	Batch	Batch or continuous process
Scrap rate	High	Potentially low (can be recycled)

Unlike thermosets, thermoplastics need only be heated to the melting point, consolidated, and then immediately cooled, a forming cycle which can result in high productivity and makes low unit cost possible. Therefore, automated equipment and continuous processing may be utilized during forming of thermoplastics.

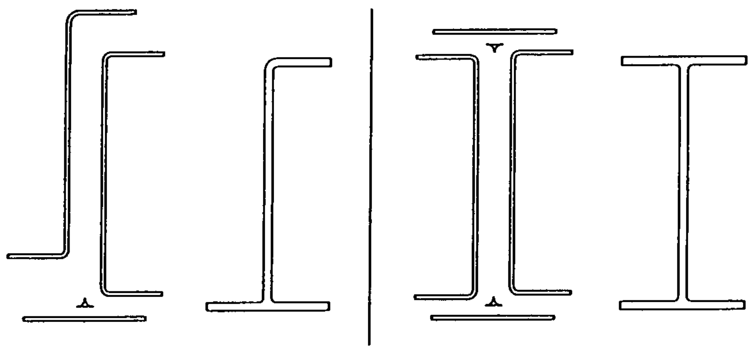
Tool Selection

The most important key to a successful fabrication program for composite structures rests on the tooling. When selecting tools for composite manufacturing, there are several factors which should be considered and usually one is of major importance, although others may also be crucial. Tool design and requirements were discussed in Chapter 3; and consideration will now be given to selecting tooling to facilitate fabrication of composite components. Several important factors are:

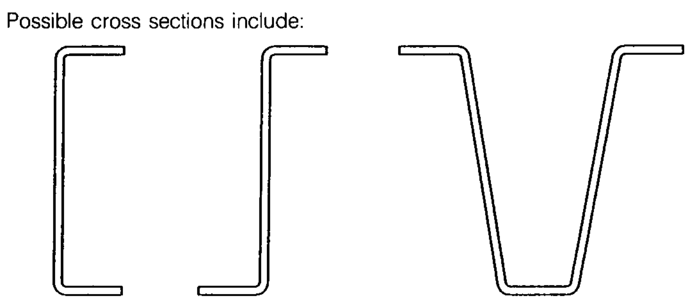
(1) Part configuration, as shown in Fig. 4.1.3, and cure or consolidation are the primary factors in tool selection.
(2) To determine the most important functions of the tool, rate the relative importance:
- Cost
- Repeatability
- Durability
- Potential for further development

Whether the tool is intended to make a prototype part or production parts and whether it is intended to make demonstration parts or be used for flight test purposes are the most important factors in tool selection.

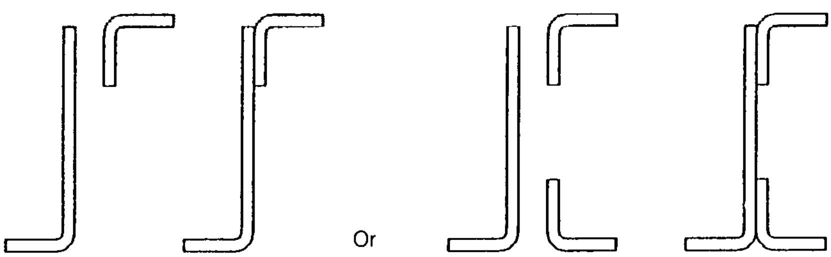
(3) Many factors must be considered when choosing a tooling system:
- Laminate material to be used



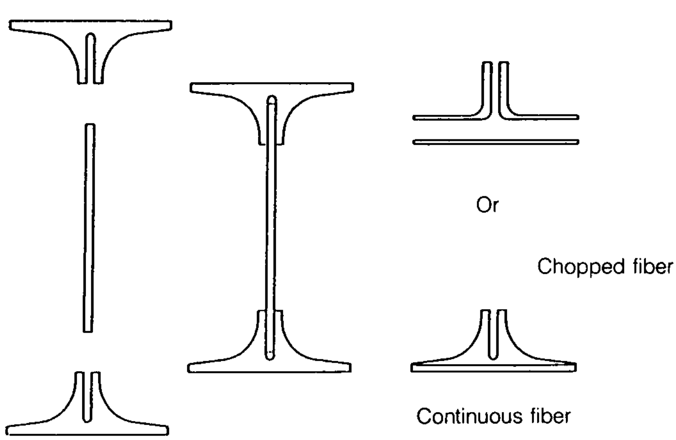
(a) Maximum tooling concepts



(b) Medium tooling



(c) Minimum tooling concepts



(d) Minimum tooling concepts

Fig. 4.1.3 Part Configuration Affects Tool Selection

- Vacuum system requirements
- Tolerance requirements
- Effect of thermal expansion
- Bonding of mating surfaces
- Strength and stiffness requirements of the tool while minimizing its bulk and heat capacity
- Size of the composite component being cured or consolidated

(4) For high temperature curing/consolidation of composite (e.g., thermoplastics), the prime concerns in tool selection are:
- Heat-up rate
- Durability
- Thermal expansion

(5) Selection of male vs. female tooling systems — This is a very important consideration in selecting a tooling system for fabricating composite structures, especially for large components. Fig. 3.1.2 in Chapter 3 shows the comparisons between the two. The male tool provides assembly advantages that may be hard to comprehend or envision unless work has been performed on both systems.
If tools are designed without considering the processing requirements of the tool/prepreg system as a whole during cure, the following may result:
- Prepreg problems conforming to the tool, resulting in bridging, or unequal buildups may form and cause pressure gradients, causing the part to warp as it cures
- Porosity and wrinkles in the finished part

(6) The tool and the work instructions should be "tried out" and any needed changes be accomplished before release to production.

Thermal Expansion Effects

Thermal expansion in composite manufacturing becomes a significant factor even at the conceptual design stage because:

- The CTE of composites are directional and vary with fiber orientation (see Fig. 4.1.4)
- Many composites have a coefficient of thermal expansion (CTE) near zero in the fiber direction and stresses may be induced where composites are joined to metals

Thermal expansion consideration include:

- Particular attention must be paid to structures which are bonded, cocured or co-consolidated and which will experience high or low service temperature
- Stresses or deflections can be induced in structures consisting of materials which have very dissimilar CTEs
- Hybrid laminates, consisting of two or more types of material in the composite layup, may experience detrimental thermal expansion effects

Methods of reducing thermal expansion effects:

- Symmetrical laminates (see Fig. 4.2.8) should always be used to minimize warpage and deflections within a laminate

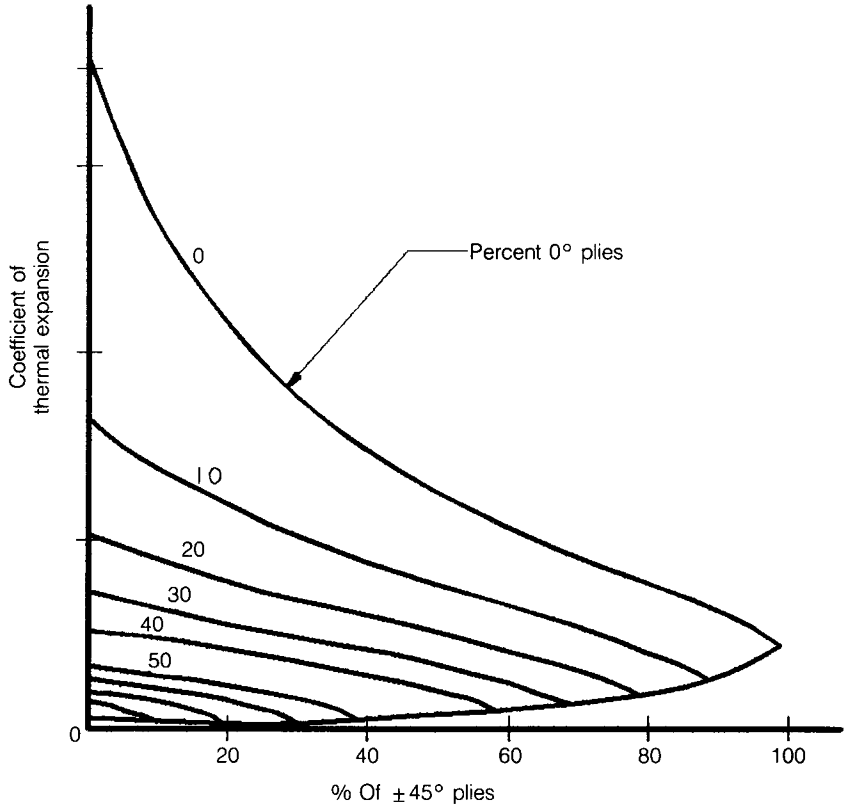


Fig. 4.1.4 CTE Varies with Percentage of 0° plies

- CTE may be increased in the 0° direction by increasing the percentage of plies in the 90° direction of a laminate. Conversely, this reduces the CTE in the 90° direction
- Tailor the laminates give the required CTE
- Metal matrix composites and aluminum (due to its greater CTE) must be given special consideration; the CTE of boron is closer to that of aluminum

Processing

Process selection is dependent on part configuration, design requirements and manufacturing capability. Before final judgments can be made as to the preferred process method, the whole system, including pros and cons should be clearly understood.
Consideration is given to:

- Process specifications — General or testing reference documents
- Process bulletins — project or individual
- Cure or consolidation cycle control:
  - — Vacuum debulking
  - — Definition of bleeder system (if required)
  - — Cure or consolidation definition: heat-up, cool-down rates, dwell cycles, cure or consolidation temperature and time, pressure, vacuum, etc.

- Material control
- Tooling
- Manufacturing procedure and controls in sufficient detail to ensure part conformance to engineering requirements
- Pilot part — initial part fabrication prior to production to verify process
- Workmanship
- Cure or consolidation cycle control
- Records:
  - Vacuum
  - Autoclave pressure
  - Temperature records based on use of multiple thermocouples on the part as defined in the specification
- Quality assurance (Q.A.) requirements and procedures
- Destructive testing requirements

The process of cure/consolidation generally involves

- Viscosity of resin decreases as heat is applied
- Tool expansion
- Vacuum is applied to remove volatiles
- Pressure is applied and laminate cure/consolidation begins
- Vacuum is vented (released)
- Resin gels and part geometry is set
- Heat is continued until resin is fully cured
- Part and tool cool down
- Tool shrinks — it could damage or trapped part

There are numbers of fabrication processes for composite manufacturing and they are identified by the facility use to provide the cure cycle or the type of tool used. It must be kept in mind that some processes will result in increased production costs and some less frequently used processes have merit for unique designs. It is fundamental to realize that there is no unique cure/consolidation cycle even for any one particular resin formulation.

Solvents and water, as absorbed moisture (usually only about 1% for thermoset and .2% for the thermoplastic of prepreg weight), are vaporized into volatiles and along with the air trapped in the layup, must be removed under high temperature and pressure processing. High pressure is usually not applied till well into the cure cycle, after most of the volatiles have escaped. If the pressure is applied prior to the removal of volatiles and air, it is entirely possible for the pressure to effectively seal off local highly curved areas and prevent the trapped vapors from escaping.

Porosity in composite parts:

- Porosity is microscopic interfacial voids dispersed throughout the thickness of a laminate
- Porosity is caused by:
  - Low resin content
  - Failure to remove volatile matter
  - Failure to compact/consolidate the laminate

- Effects of porosity: (> 2%):
  - Slight reduction in static compression strength
  - Significant reduction in static interlaminar shear strength
  - Significant reduction in interlaminar shear fatigue strength
  - No reduction in compression fatigue strength

The following should be considered when selecting a manufacturing process:

(1) Composite material and form selections will greatly influence the processing choice; prepregs are usually used because:
- Elimination of formulation problems and the mess of wet-layup
- Consistent quality and resin content
- Adaptability to assembly line technique for increased production at lower unit costs
- Fewer rejects
- Less variance in mechanical properties
- Best quality materials
- Design freedom because of the ease with which prepregs can be formed into irregular shapes
- Lower inventory levels since no resins or catalysts need be stocked

(2) Tool selection should be one of the prime factors in process selection.

(3) The tightness of tolerances and which surfaces of the laminate part are controlled often exert a strong influence on the selection process. Tight tolerances almost invariably lead to additional expense and the control surfaces have a strong influence on whether male or female tools are used.

(4) Always consider the option of making a single relatively complicated part to perform the same function as a number of simpler components which must be joined. In addition to increasing structural efficiency, this results in a reduction in planning and tool costs, because one complex tool will usually not cost as much as a greater number of simpler tools. This in turns reduces production control, tool inventory and therefore total production costs.

(5) Cost reduction is possible when secondary bonding and/or mechanical fastening is eliminated. However, this approach must be balanced by operational requirements and need for future repair or replacement of damaged parts.

(6) More automatic computer control autoclaves would increase consistency and reduce operator induced variances. Statistical process control data should be collected for traceability, analysis and historical record as well. Cure cycle data could be optimized and downloaded to the autoclave system.

(7) Processability trials
- Many composite manufacturing specifications require fabrication of a trial part representing the production design
- Fabrication of a trial part will determine if the design, material and process can be used for production parts

Possible manufacturing processes include:

(1) Autoclave
- Costly — US$3000 to 10,000 per run depending on part size and complexity

Next Page

- Requires compatible cure or consolidation process systems (temperature and pressure)
- Requires compatible tool thermodynamics
- Long cycle times

(2) Oven
- Much lower cost than autoclave
- Usually requires trapped rubber tooling to provide pressure
- Shorter cycle time than autoclave
- Low pressure capability

(3) Press forming
- Very short cycle time
- Good dimensional control since inner and outer mold lines are controlled
- Common net resin system
- Net shape parts

Manufacturing Cost

Besides the structural weight savings, manufacturing cost has become as important as performance in determining the success or failure of a particular airframe design. The factors involved in controlling cost are:

- The use of accurate, realistic cost estimating tehcniques
- Identification and resolution of potential cost problems early in the design phase, avoiding large cost overruns or schedule slippage
- Manufacturing process selection
- Material choices
- Geometry complexity
- Assembly process
- Layup process

Several guidelines are useful when trying to estimate actual costs:

- Know what drives the cost of a particular design
- Optimize the configuration for minimum cost while still meeting performance goals
- Determine the actual cost using available estimation techniques and historical data (eliminate conservatism in cost estimation)
- Compare cost target
- Use of the comparison of material placement costs shown in Fig. 4.1.5

Safety and Health

Many of the chemicals used in the manufacture of composites may be harmful and/or toxic (see Ref. 1.2 and Ref. 4.60); therefore care should be execised when using these materials. Some basic points to consider are:

- The means to better monitor and control the clean room environment must be obtained to meet specifications and improve continuity of composites fabrication processes

Chapter 5.0

JOINING

5.1 INTRODUCTION

A complete airplane structure is manufactured from many parts such as skins, stiffeners, frames, spars, etc. These parts must be joined together by fastening, bonding, welding, etc., to form subassemblies are joined together to form larger assemblies and then finally the completed airplane. Many parts of the completed airplane must be arranged so that they can be disassembled for shipping, inspection, repair, or replacement. Fasteners are usually used to join such parts. In order to facilitate the assembly and disassembly of the airplane, it is desirable for such fastened joints to contain as few fasteners as possible. Fig. 5.1.1 gives a comparison of joining methods.

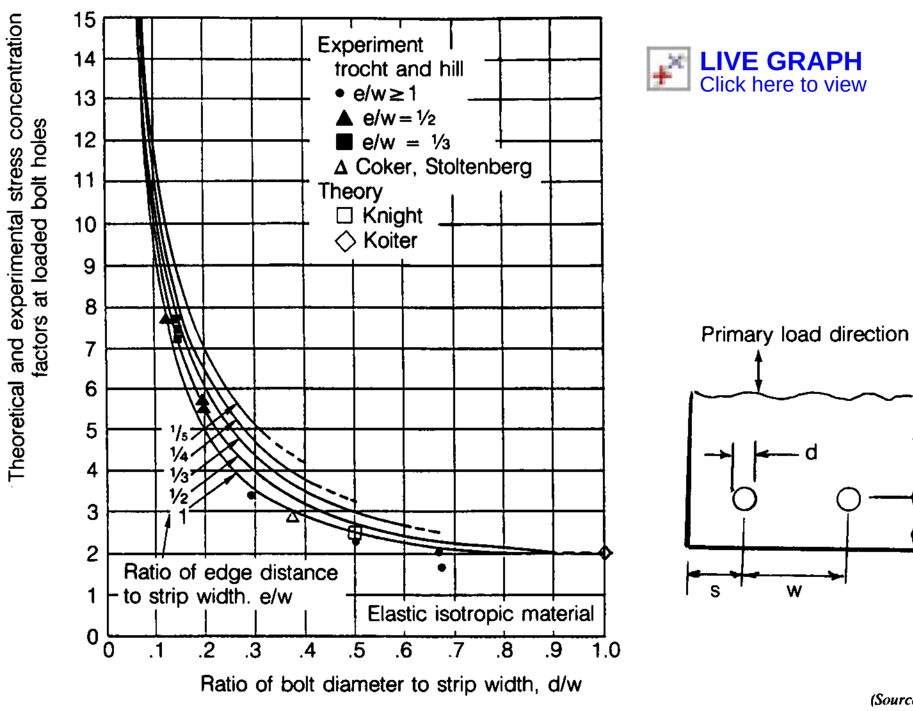
Joints are perhaps the most common source of failure in aircraft structure and therefore it is most important that all aspects of joint design are given consideration during the structural design. Failures may occur for various reasons, such as secondary stresses due to eccentricities, stress concentrations (especially for fastened joints as shown in Fig. 5.1.2) excessive deflections, etc., or some combination of conditions, all of which are difficult to evaluate to an exact degree. These factors directly affect the strength of joints, especially the fastened joints which are greatly weakened by notch effects.

Assembly joints, which occur when any two components are assembled, are a major source of stress concentrations. In the case of bonded joints, stress concentrations occur to maintain strain compatibility between bonding components. In the case of mechanical joints, they are a result of the decreased area at the hole and the loaded hole itself. The primary purpose of this section is to acquaint the engineer with some of the problem areas encountered, introduce some of the joint design allowables generated on the subject, and show a few examples of how typical problems have been solved.

To fully realize the potential of advanced composites in lightweight aircraft structure, it is particularly important to ensure that the joints, either bonded or fastened, do not impose a reduced efficiency on the structure. This problem is far more severe with composite materials than with conventional metals because the high-specific-strength composite filaments are relatively brittle. Composites have very little capacity to redistribute loads as shown in Fig. 5.1.3 and practically none of the forgiveness of a yielding metal to mask a multitude of design approximations. This is the reason why greater efforts are devoted to understanding joints in composite materials and to providing reliable design techniques, particularly for thicker sections and for multiple fastener pattern design cases.

Method		Anticipated Benefits	Limitations
Mechanical fastening		• Mature Technology • Baseline for cost data • Could supplement weld/bond assembly methods	• Low risk • Increased weight • Labor Intensive • Requires secondary seal • Shimming fit-up stress
Adhesive bonding		• Reduced fastener count/weight	• Moderate risk • Cure cycle required • Tooling
Thermoplastic Welding	• Resistance	• Can be automated process • Continuous weld • Reduced fastener count/weight	• Moderate risk • Requires 2 side access
	• Ultrasonic	• Can be automated process • Possible continuous weld • Reduced fastener count/weight	• Moderate risk • Requires 2 side access
	• Induction	• Requires 1 side access • Can be automated process • Continuous weld • Reduced fastener count/weight	• Moderate — high risk • Requires magnetic susceptor mat'l
Cocuring		• Total homogeneous weld joint • Probable elimination of seal	• Low risk • Part size/shape limited

Fig. 5.1.1 Comparison of Joining Methods



(Source: Ref. 5.4)

Fig. 5.1.2 Stress Concentration Levels Rise Rapidly for Fastener Holes Smaller Than 0.2 Times The Strip Width

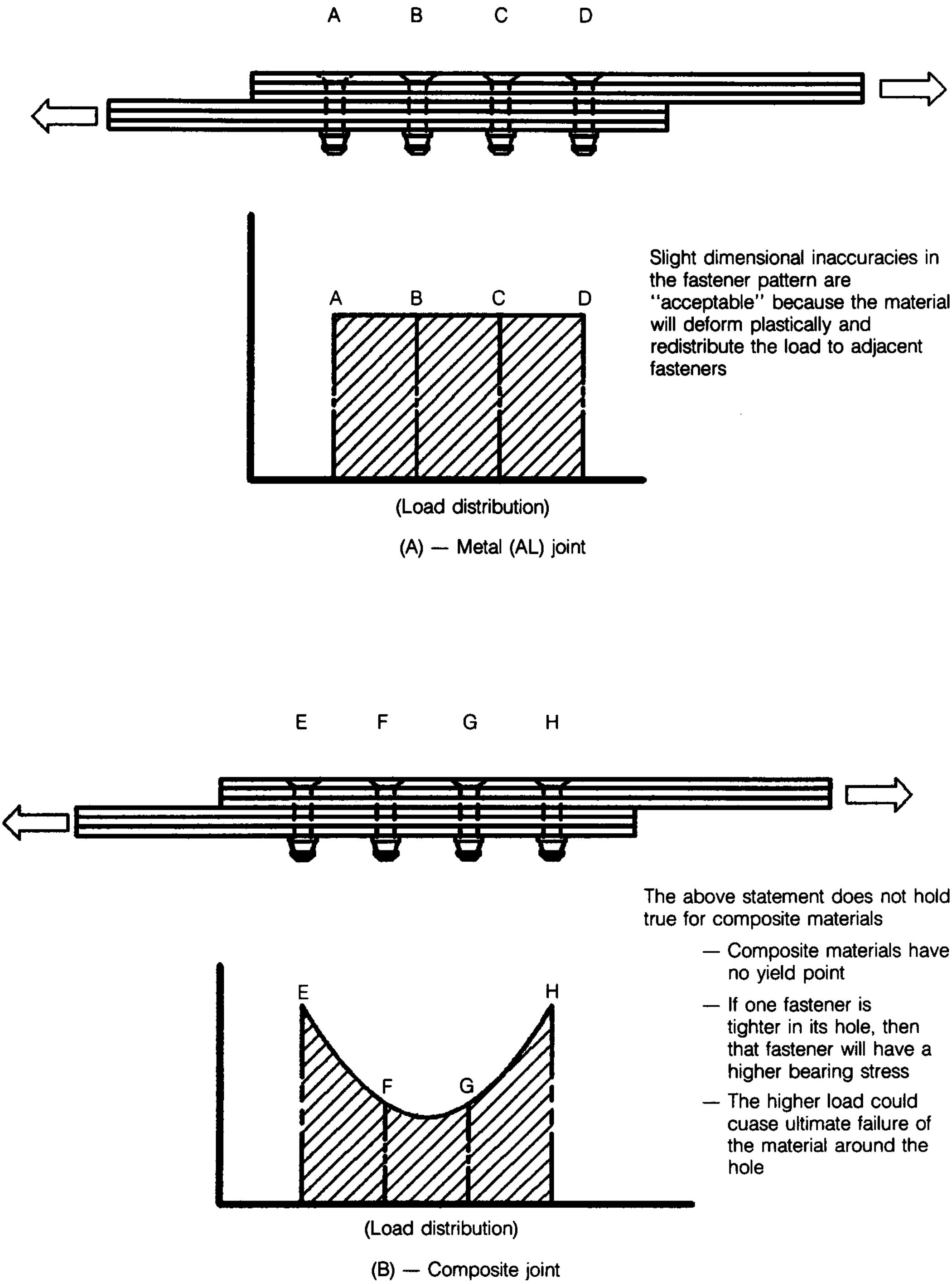


Fig. 5.1.3 Comparison Between Metal and Composite Joints

There are six basic factors to be considered in the design of a composite joint:

- The loads which must be transferred
- The region within which this must be accomplished
- The geometry of the members to be joined
- The environment within which the joint must operate
- The weight/cost efficiency of the joint
- The reliability of the joint

The first four items are generally prescribed and it remains then to satisfy the last two items in some optimal manner. The first decision should be to select the class of joining techniques which should be studied. In the past, structural engineers considered adhesive joints to be more efficient for lightly-loaded joints, while mechanically fastened joints were thought to be more efficient for highly-loaded joints. This is definitely not always the case with composites. The notch sensitivity of composite materials at fastened joints greatly reduces the joint efficiency in composite structural assembly, as shown in Fig. 5.1.4. To realize maximum efficiency from adhesives, joints should be specifically designed for adhesive bonding or cocuring/co-consolidation methods. Adhesively bonded or co-cured joints can overcome many of these limitations.

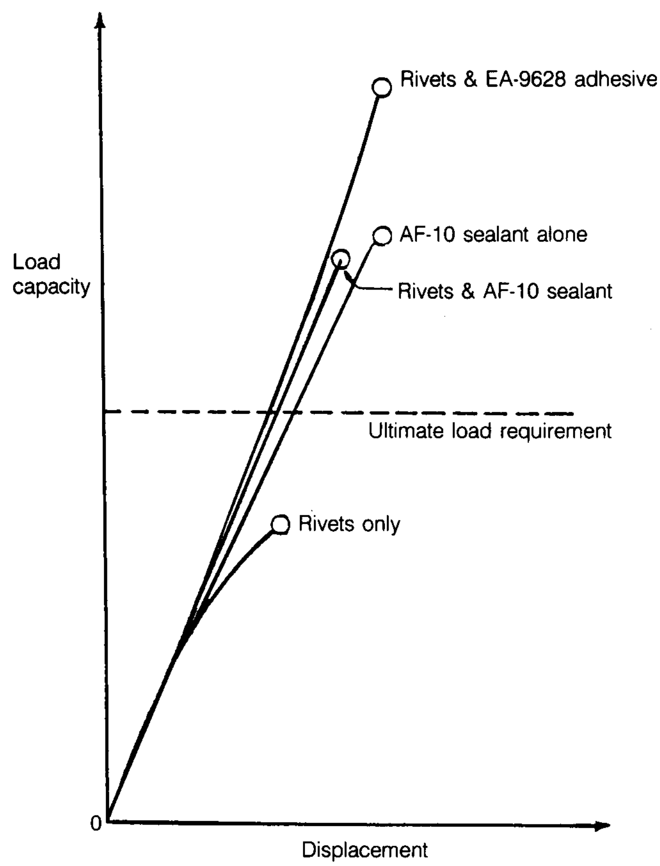
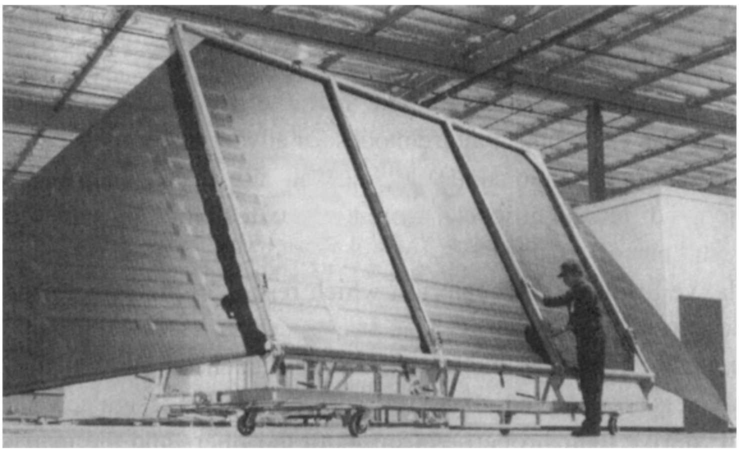


Fig. 5.1.4 Inadequacy of Rivetted Splices (Used on Lear Fan Fuselage)

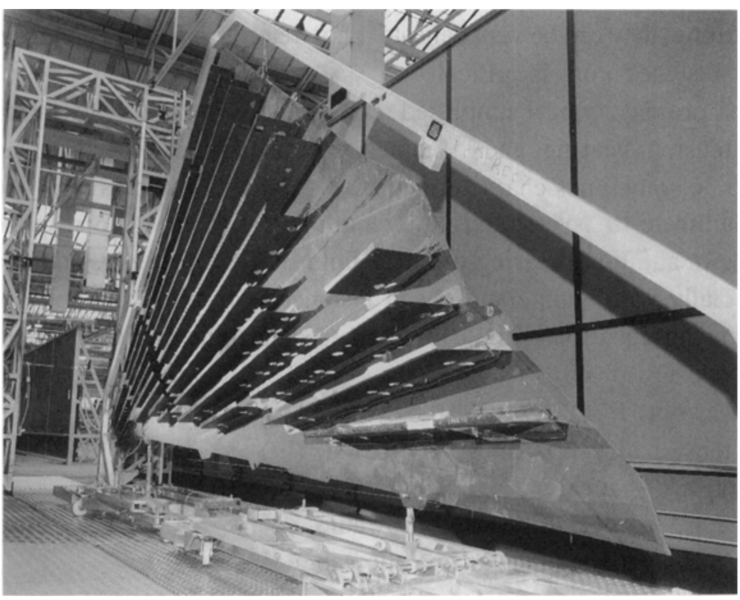
It is a general rule that the most efficient composiste joints are scarf and stepped lap joints in which there is relatively little change in the load path. Double-lap and single-lap joints are quite a bit less efficient, in that order.

To achieve the goal of both structural weight reduction and cost savings, as many mechanically fastened joints and/or bonded joints as possible must be eliminated. Fig. 5.1.5 shows an example of an one piece composite wing skin panel for a fighter aircraft on which wing spanwise splice and skin-spar joints were eliminated.



By courtesy of The Boeing Co.

(a) One piece thermoplastic wing skin.



By courtesy of British Aerospace (Military Aircraft) Ltd.

(b) EFA wing spars co-bonded to the lower skin surface

Fig. 5.1.5 UseOne Piece and Co-bonded Method to Eliminate Fasteners

In the end there will be some requirements for mechanical fasteners to hold access doors, removable covers (e.g., a fighter wing upper cover which must be removed for maintenance purposes), detachable parts, etc., or to join components in final assembly. Since composite materials have some unique properties and characteristics, most fasteners selected for joining composites are tailored to the these materials to avoid problems.

Because of the many factors and unknowns involved in designing composite laminate joints, tests should be conducted which simulate the operational environment and loading in order to insure joint reliability.

5.2 MECHANICAL FASTENING

The use of mechanical fasteners to assemble airframe structures is a mature technology. Composites are not an exception. Failure modes for advanced composite mechanical joints are similar to those for conventional metallic mechanically fastened joints. But the behavior of composite joints differs significantly from those of metallic joints and deserves special attention of a number of reasons:

(a) Relative brittleness of material, which results in high stress concentrations at hole edges

(b) Laminate failure is a function of stacking sequence, fiber volume, porosity, etc.

When fasteners are required, composites present special design considerations. Composite materials derive their properties from both the fibers and the matrix, and are not homogeneous. They do not respond to fasteners in the same way as metals. Therefore, it is not possible to design fasteners that are universally applicable to all composites. Composites possess different characteristics than their aluminum counterparts: even though they are very strong, they can be very delicate if not treated properly. Therefore, the selection of the correct fastened joint is critical.

Fig. 5.2.1 presents typical simplified representations of the following failure modes: shear out, net tension, bearing, and combined tension and shear out. The shear out failure mode can also be sometimes characterized by a single-plane "cleavage" failure, where the apparent laminate transverse tensile strength is less than the corresponding in-plane shear strength. In addition, bearing or shear failure of the fastener, and fastener pulling through (especially with countersunk head fasteners) are other possible failure modes.

The following equations should be used to determine allowable joint strengths:

- Bearing: $p^{br}=(d)\ (t)\ (F^{br})$
- shear out: $P^{s}=2[(e/d)\ -\ 0.5]\ (d)\ (t)\ (F^{s})$
- Net tension: $P^{t}=2[(s/d)\ -\ 0.5]\ (d)\ (t)\ (F^{t})$

where 0.5 — inch

F^{br} — design bearing allowable

F^{s} — design shear out allowable

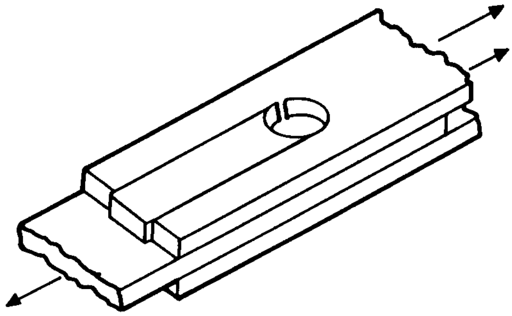
F^{t} — design net tension allowable

d — fastener diameter

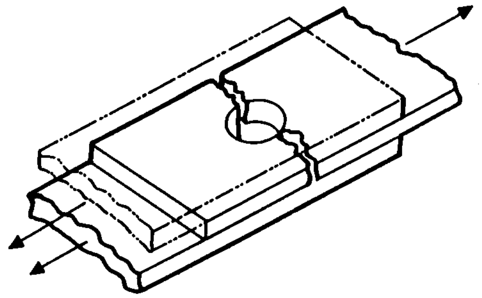
t — Laminate thickness

e — edge distance

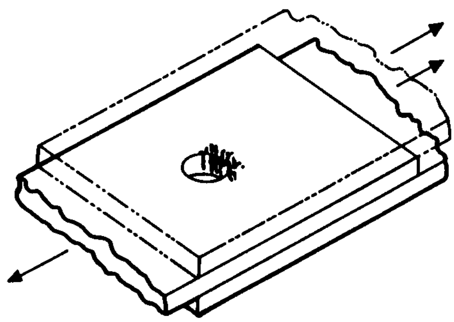
s — fastener side distance (see Fig. 5.1.2)



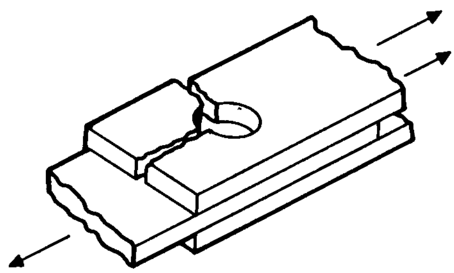
(a) Shearout failure



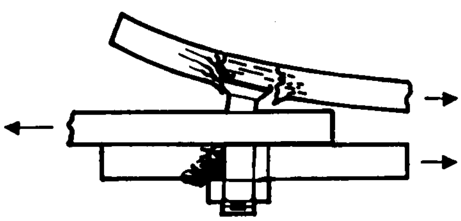
(b) Tension failure



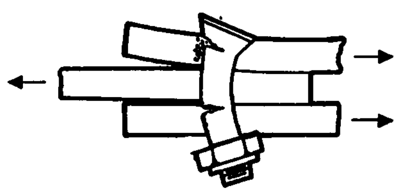
(c) Bearing failure



(d) Cleavage — tension failure



(e) Bolt pulling through laminate



(f) Bolt failure

Fig. 5.2.1 Fialure Modes of Advanced Composite Mechanical Joints.

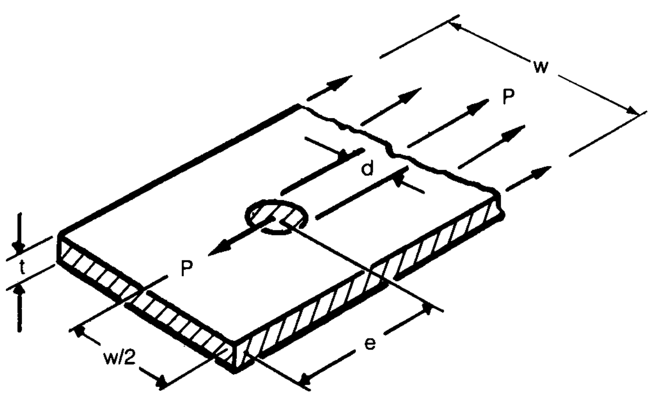


Fig. 5.2.2 Typical Mechanical Joint Element.

The equation given above for predicting shear out strength utilizes an equation applicable for both shear out and cleavage failure, since F^s has been empirically obtained to cover both cases. Fig. 5.2.2 represents an element of a typical mechanical joint and defines the key dimensions by illustration. Although many assembly problems have been solved with adhesive-bonding techniques, there are many cases where only mechanical joints are capable of meeting design requirements. Example include parts requiring replacement or removal for ease of fabrication or repair.

Some of the obvious advantages of mechanical joints are:

- Utilization of conventional metal-working tools and techniques, as opposed to adhesive-bonding procedures
- Ease of joint inspection
- Utility of repeated assembly and disassembly for fabrication replacement, or repair
- Assurance of structural reliability
- Little or no surface preparation or cleaning required
- Easily inspected for joint quality

Offsetting the advantages are some concerns, which are:

- Strength degradation of the basic laminate notched effect and a resultant weight penalty since a local buildup thicker laminate is needed to offset the notched effect
- The need for more careful design than used with conventional metals because of the lack of ductility to relieve local stress concentrations and because of the unequal directional properties of the laminate
- Possible increased cost because of increased number of operations required
- Potential fastener corrosion resulting from contact with carbon composite materials

The following design practices are recommended for mechanically fastened joints:

- Stress concentrations exert a dominant influence on the magnitude of the allowable design tensile stresses. Generally, only 20-50% of the basic laminate ultimate tensile strength is developed in a mechanical joint
- Mechanically fastened joints should be designed so that the critical failure mode is in bearing, rather than shear out or net tension, so that catastrophic failure is prevented. This will require an edge distance to fastener diameter ratio (e/d) and a side distance to fastener diameter ratio (s/d) relatively greater than those for conventional metallic materials. At relatively low e/d and s/d ratios, failure of the joint occurs in shear out at the ends , or in tension at the net section. Considerable concentration of stress develops at the hole, and the average stresses at the net section at failure are but a fraction of the basic tensile strength of the laminate
- Multiple rows of fasteners are recommended for unsymmetrical joints, such as single shear lap joints, to minimize bending induced by eccentric loading
- Local reinforcing of unsymmetrical joints by arbitrarily increasing laminate thickness should generally be avoided because the resulting eccentricity can give rise to greater bending stress which counteracts or negates the increase in material area

- Since stress concentrations and eccentricity effects cannot be calculated with a consistent degree of accuracy, it is advisable to verify all critical joint designs by testing a representative sample joint

Mechanical Joint Design Guidelines

In general, the best fastened joints in fibrous composites still impose a loss in strength of about half the basic material allowable (although there is test evidence to indicate that interference-fit fasteners can alleviate this reduction toward a net area loss):

(1) If a laminate is dominated by 0° fibers with few 90° fibers it is most likely to fail by shear out. Unlike metals, in which shear out resistance can be increased by placing the hole further from the edge, laminates are weakened by fastener holes regardless of distance from the edge. Reinforcing plies at 90° to the load helps prevent both shear out [see Fig. 5.2.1(a)] and cleavage [see Fig. 5.2.1(b)] failures:
   - Use larger fastener edge distances than with aluminum design, such as e/d > 3
   - Use a minimum of 40% of ±45° plies; see Fig. 5.2.3 for the effect of layup on the bearing stress at failure
   - Use a minimum of 10% of 90° plies

(2) Net tension failure [see Fig. 5.2.1(b)] is influenced by the tensile strength of the fibers at fastened joints, which is maximized when the fastener spacing is approximately four times the fastener diameter (see Fig. 5.2.4). Smaller spacings result in the cutting of too many fibers, while larger spacings result in bearing failures, in which the material is compressed by excessive pressure caused by a small bearing area:
   - Use minimum fastener spacing as shown in Fig. 5.2.5
   - Pad up to reduce net section stress

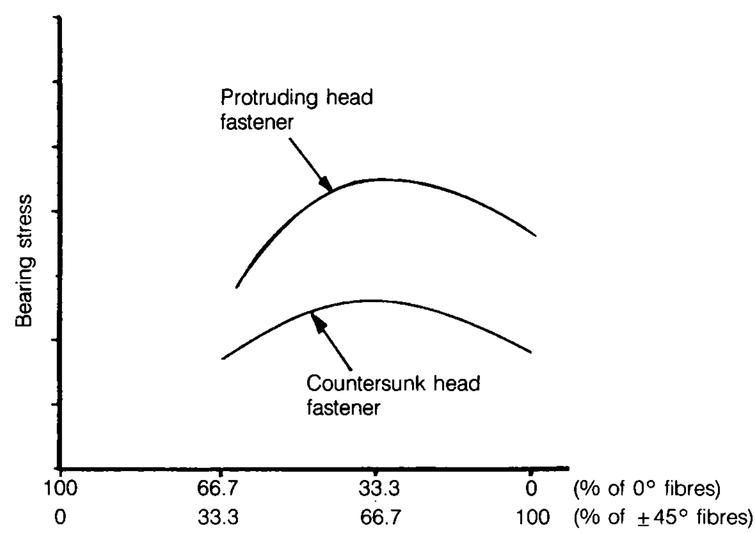


Fig. 5.2.3 The Effect of Layup on the Max. Bearing Stress.

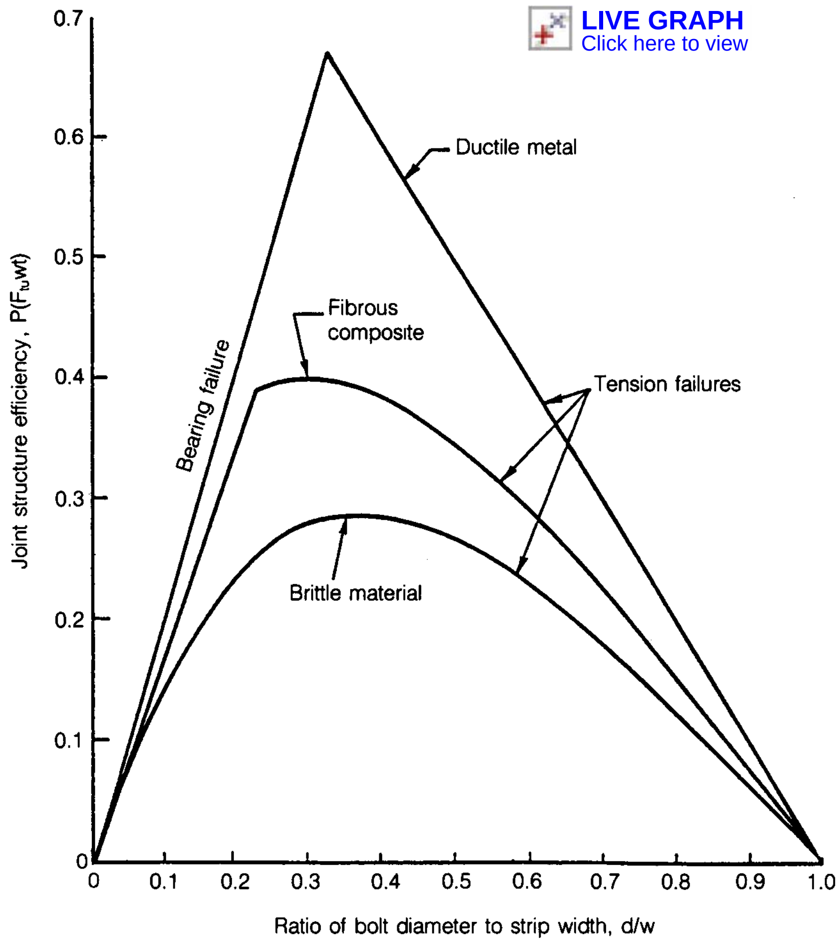


Fig. 5.2.4 Relation Between Strengths of Fastened Joints in Ductile, Brittle, and Composite Materials

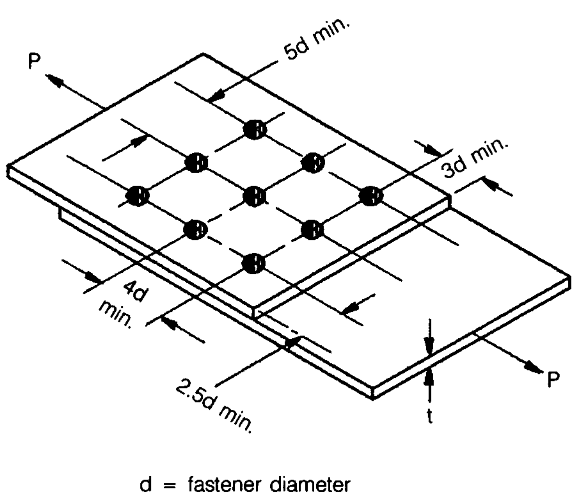


Fig. 5.2.5 Minimum Fastener Spacing and Edge Distance

(3) Fastener pull-through from progressive crushing/bearing failure [see Fig. 5.2.1(c)]:
- Design joint as critical in bearing
- Use padup
- Use a minimum of 40% of $\pm 45°$ plies
- Use washer under collar or wide bearing head fasteners
- Use tension protruding heads when possible

(4) Fastener shear failure:
- Use large diameter fastener
- Use higher shear strength fastener
- Never use a design in which failure will occur in shear

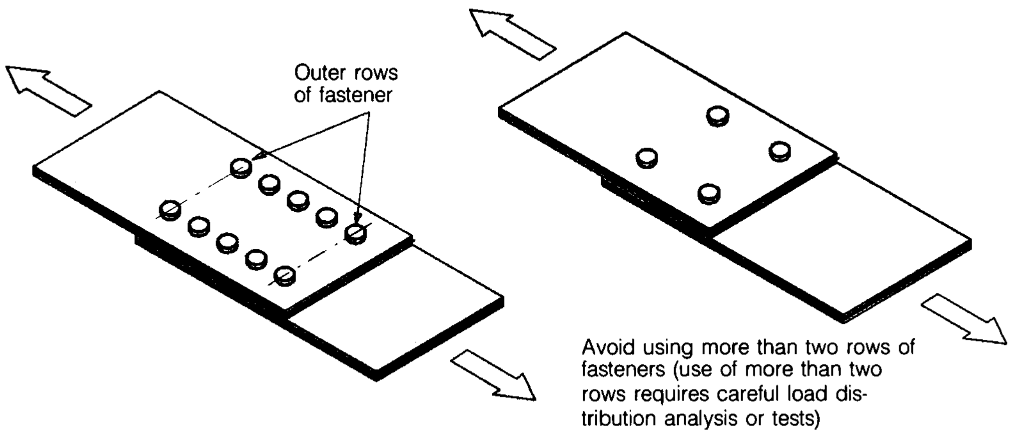
(5) Use two row joints when possible. The low ductility of advanced composite material confines most of the load transfer to the outer rows of fasteners (see Fig. 5.2.6)

(6) The use of carbon composites in conjunction with aircraft metals is a critical design factor. Improper coupling can cause serious corrosion problems for metals. Materials such as titanium, corrosion-resistant steels, nickel and cobalt alloys can be coupled to carbon composites without such corrosive effects. Aluminum, magnesium, cadmium plate and steel will be most adversely affected because of the difference of electrical potential between these materials and carbon. Fig. 5.2.7 shows the galvanic compatibility of fastener materials with carbon composites.

(7) The choice of optimum layup pattern for maximized fastener strength is simplified by the experimentally established fact that quasi-isotropic patterns $(0/\pm 45/90)_s$ or $(0/45/90/-45)_s$ are close to optimum. This reduces experimental costs and simplifies the analysis and design of most fastened joints.

(8) One of the key factors governing fastened joint behavior in advanced composite structure is the vast difference between double-lap and single-lap joint efficiencies. The eccentricity in the load path for single-lap joints leads to non-uniform bearing stresses across the thickness of the laminate (see Fig. 5.2.8). This, in turn, leads to the development of the critical bearing stress and bypass stress around hole at the laminate interface at an even lower than average bearing stress because of the brittle nature of composite materials. Fig. 5.2.9 shows bearing stress distribution at the fastener hole, and the use of the bearing reduction factor to account for this effect. It is difficult to define the reduction factor because it is a function of fastener material characteristics, composite material and layup sequence, fastener fit, etc. Currently, an arbitrary value of 1.5 to 2.0 is used for the reduction factor until results from testing are established for each particular design case.

(9) Develop a bearing/bypass stress interaction envelope curve (function of laminate material, laminate thickness and ply layup sequence or tacking, fastener diameter, etc.) to size mechanical joints as shown in Fig. 5.2.10.



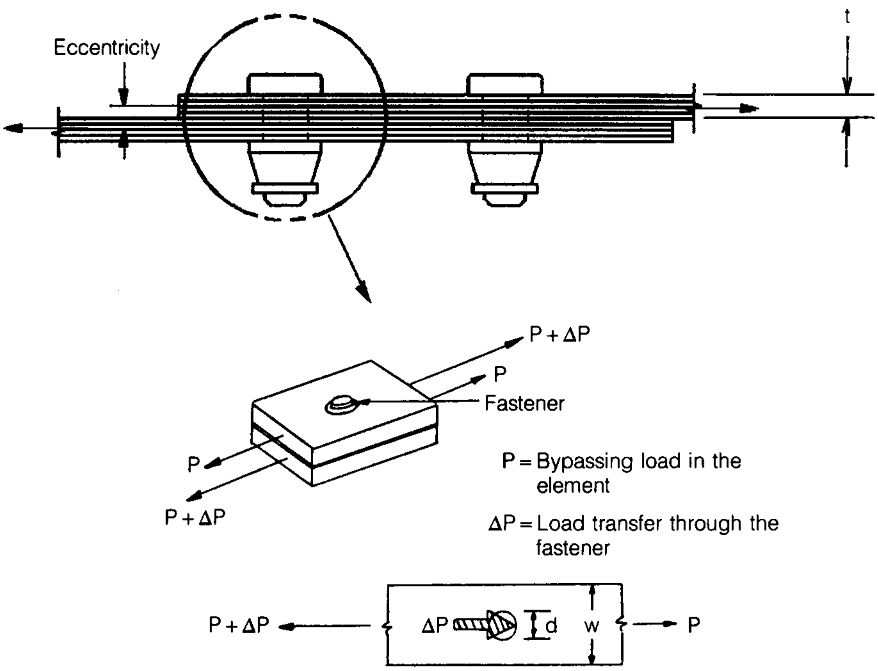
(a) Multi-row joint (avoid)

(b) Two-row joint (prefer)

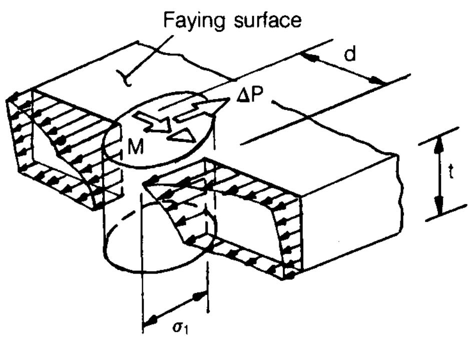
Fig. 5.2.6 Outer Rows of Fastener Carry Most of Load Due to The Low Ductility of The Composite Materials

Fastener Material	Compatibility with Graphite/Epoxy and Application Guidelines
Titanium, Ti Alloys, Ti-CP	Fasteners of these materials are compatible with graphite/epoxy composites. Permanent fasteners should be sealed to prevent water intrusion but removable fastener may be used with no supplement protection.
MP-35N (AMS 5758) Inco 600 (AMS 5687)	These materials are compatible with graphite/epoxy components.
A286 (AMS 5731, AMS 5737) PH13-8Mo (AMS 5629)	These CRES alloys and some other austenitic and semi-austenitic alloys are marginally acceptable in contact with graphite composites. In a severe marine/industrial corrosion environment, superficial rusting and stains develop on the fastener. Although loss of fastener integrity has not been established, this staining is usually objectionable. Permanent fasteners that can be installed with sealant and overcoated with sealant are usually satisfactory. Removable fasteners are not acceptable to some design activities.
Monel	Marginally acceptable in contact with graphite/epoxy composites. Significant current flow and material loss.
Low Alloy Steel, Martensitic Stainless Steels	Not compatible with graphite/epoxy materials. Severe rusting.
Silver Plate, Chromium Plate, Nickel Plate	These plating materials are compatible with graphite but are not adequate to protect steel in contact with graphite/epoxy composites. Silver plated A286 or PH13-8Mo would be compatible with graphite and suitable if there is no aluminum or titanium in the joint.
Cadmium Plate, Zinc Plate, Aluminum Coatings	Not compatible with graphite/epoxy composite materials. Rapid deterioration of plating or coating.
Aluminum, Aluminum Alloys, Magnesium Alloys	Not compatible with graphite/epoxy composite materials. Generally, it is feasible to adequately protect fasteners of these materials from severe corrosion if in contact or close proximity to graphite.

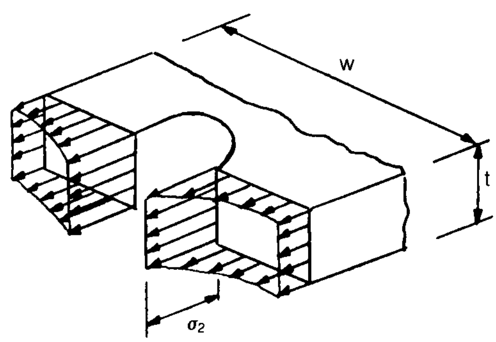
Fig. 5.2.7 Galvanic Compatibility of Fastener Materials with Graphite Composite.



Load fastener load transfer and bypass load.



Local stresses caused by load transfer, ΔP



Local stresses caused by bypassing load, P

The maximum local stress in the considered element,

$$\sigma_{max} = \sigma^1 + \sigma_2$$

(Note: For the σ_{max} calculation refer to chapter 7.0 of Ref. 5.30)

Fig. 5.2.8 Local Peak Stresses Caused by Load Transfer and Bypass Load

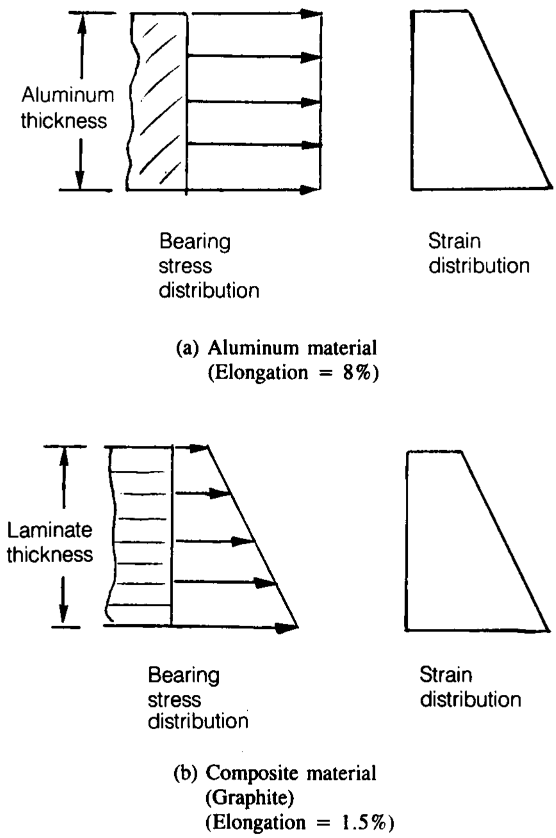


Fig. 5.2.9 Ultimate Bearing Stress Distribution of Aluminum Material Vs. Composite Material (single-lap joint).

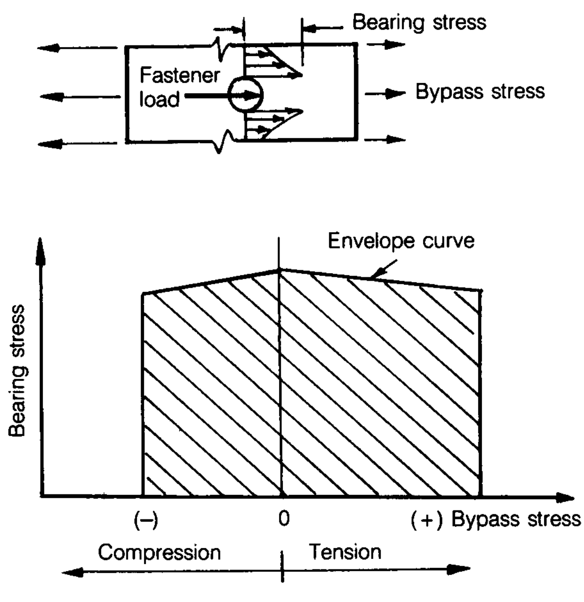


Fig. 5.2.10 Bearing/bypass Stress Interaction Envelope Curve (notched and wetted test data).

(10) Eccentricities and their effect on the joint and the surrounding structures:
- If eccentricities exist in a joint, the moment produced must be resisted by the adjacent structures
- Eccentrically loaded fastener patterns may produce excessive stresses if eccentricity is not considered

(11) Mixed fastener types — It is not good practice to employ both permanent fasteners and screws in combination in a joint. Due to the better fit of the permanent fasteners, the screws will not pick up their proportionate share of the load until the permanent fasteners have deflected enough to take up the clearance of the screws in their holes

(12) Do not use a long string of fasteners in a splice. In such cases, the end fasteners will load up first and yield early. Three, or at most four, fasteners per side is the upper limit unless a carefully tapered, thoroughly analyzed splice is used (wherever possible use a double shear splice). Study cases are shown in Fig. 5.2.11.

(13) Use tension head fasteners (potentially high bearing stress under the fastener head cause failure) for all applications. Shear head fasteners may be used in special applications.

(14) Driven fasteners, e.g., MS20470, MS20426 and DD rivets, should never be used to assemble composites

(15) If local buildup is needed for fastener bearing strength, total layup should be at least 40% $\pm 45°$ plies

(16) Install fasteners wet with corrosion inhibitor may be required

(17) Use large diameter fastener in thicker composite assemblies (e.g., to transfer critical joint loads, fastener diameter should typically be about the same size as the laminate thickness) to avoid peak bearing stress due to fastener bending. Fastener bending is much more significant for composites than for metals, because composite laminates are thicker (for a given load) and more sensitive to non-uniform bearing stresses (due to brittle failure modes)

(18) Don't buck rivets and conventional enlarged end blind fasteners

(19) The best fastened joints can barely exceed half the strength of unnotched laminate

(20) Peak hoop tension stress around fastener holes is roughly equal to the average bearing stress

(21) Fastener bearing strength is sensitive to through-the-thickness clamping force of laminates (see Fig. 5.2.12)

(22) For blind fasteners, use big-foot fasteners wherever possible

(23) Production tolerance build ups:
- Proper tolerance should be given to minimize the need for shimming
- Shim allowance should be called out on engineering drawing
- Since production tolerances can easily be exceeded in the thickness tolerance, fastener grid length is affected

(24) In fuel tank, to prevent fuel tank leakage of a groove seal design, the max. seal gap is 3 mils

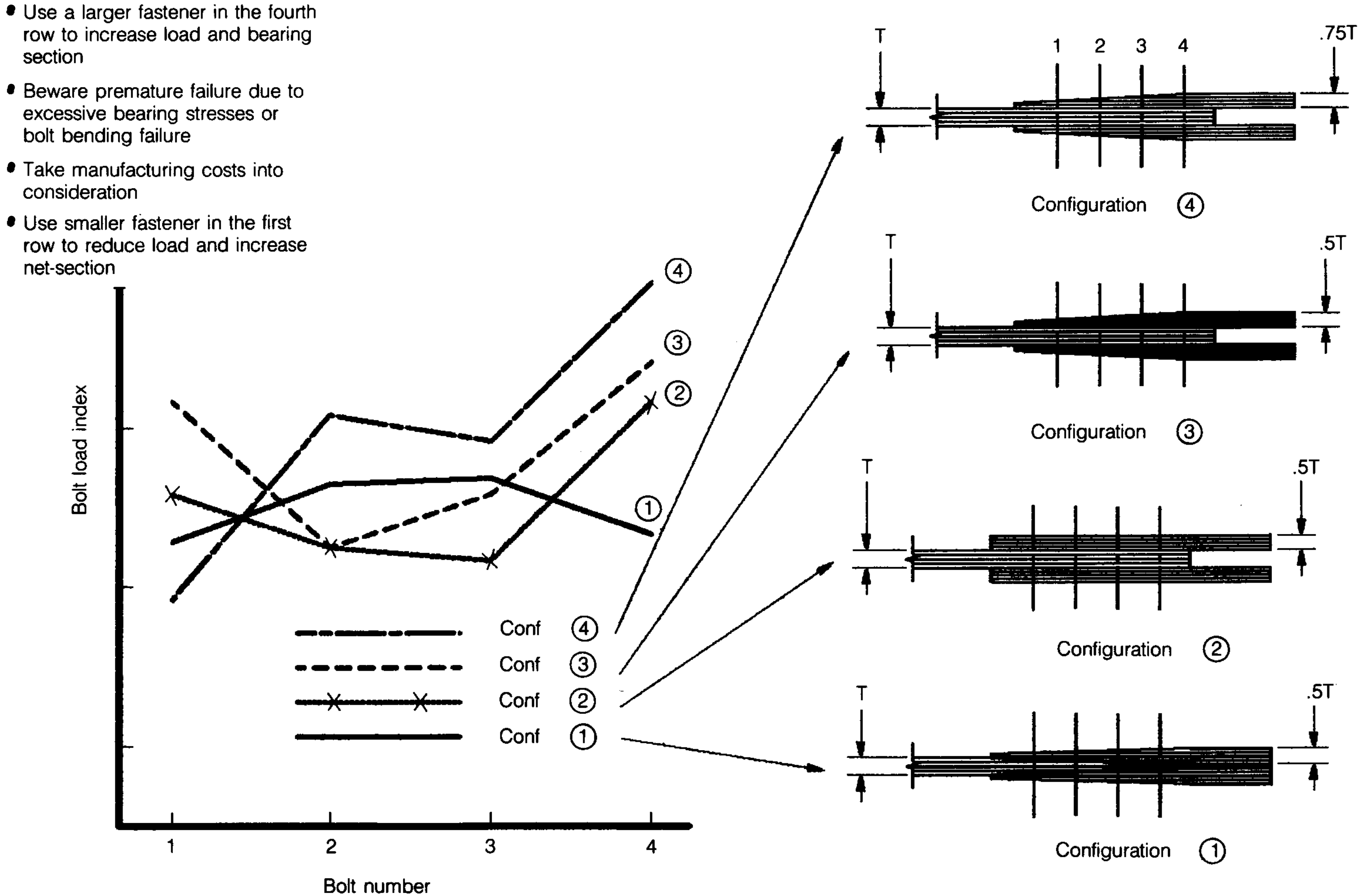


Fig. 5.2.11 Effect of Joint Configuration on Fastener Load Distribution

(25) Sparking and arcing between metal fasteners in fuel tank environments (due to lightning strike) requires special design attection (see Chapter 6)

Fasteners (Metallic)

The fasteners used with composite assemblies are generally titanium alloy, (to prevent galvanic corrosion) tension heads (to avoid fastener pull-through).

(1) Fastener materials — Fastener materials have to be environmentally compatible with the graphite laminate material to avoid galvanic corrosion; see Fig.5.2.7.
- Materials that have been found to be compatible are: titanium, inconel, and A286 steel
- Alloys which can be used with caution are: Stainless steels, monel, and PH steels
- Non-compatible alloys such as aluminums and low alloy steels may be considered if reliable coatings or environmental considerations permit their use.

(2) Fastener configuration — Composite joint strength is sensitive to fastener configuration. For primary structures, a shallow head and large diameter shear fasteners will be more efficient in thin laminates where thickness is less than one diameter. However, the standard tension head fastener is more efficient for thick laminate joints where t/d is greater than 1.0. An interference fit (e.g., sleeve fastener system) should be considered in critical joint applications.

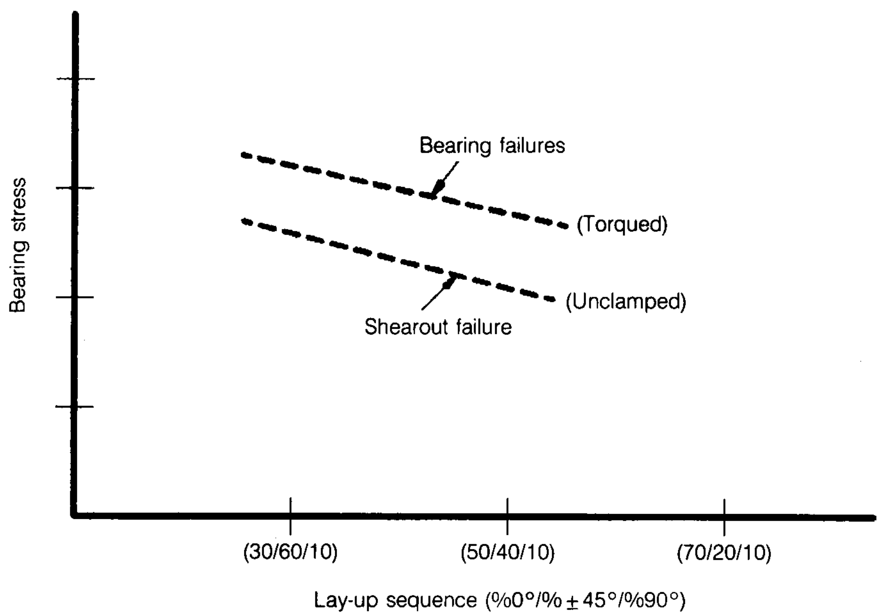


Fig. 5.2.12 Effects of Fiber Layup/Sequence Variation on Bearing Strength

Fig. 5.2.13 through 5.2.15 shows a metallic HI-LOK fastener system selected off-the-shelf to be the primary fastener for composites in the early 1970's. They are still used in many applications. These fasteners are used in clearance holes and clamp-up forces are not modified for composites. A concerned issue encountered when conventional fasteners are not modified specifically for use in composites involves clamping forces. These forces, if not controlled, can cause the fastener to crush or deform the matrix that binds the fibers together, allowing the fastener to pull through after repeated loads. Therefore, do not over-torque the fastener during installation. Fig. 5.2.16 shows a fastener systems which is similar to that of the HI-LOK fastener system except that the collar torque system is slightly different. HI-LOK uses a torqued-off collar wrenching device which Eddie bolt 2 uses the Eddie lobes for torquing and when the predetermined clamp-up force is reached the lobes deform into and acorss the pin thread flutes to form a positive mechanical lock.

Fig. 5.2.17 through 5.2.20 show the HUCK-COMP fastener which is a specially designed version of the HUCK lightweight titanium lockbolt system. It is designed specifically for composite structure applications. It is an all titanium system with a Ti-6A1-4V pin and a flanged commercially pure titanium collar. The fastener comes in both a pull system version for installation with common pull tools or an automatic drill machine version for use with DRIVMATIC machines. Fig. 5.2.21 and 5.2.22 show HUCK-TITE fastener with a sleeve which is designed to be installed with interference fits (an interference-fit range of 0.001 to 0.006 inch would be possible) in composite structure without causing any installation damage and gaining improved structural strength and tightness.

Blind fasteners are designed to be installed where access to both sides of a sheet assembly or structure is not possible or practical. For instance, skins where there is no access to the back-side such as fighter or general aviation wing covers, small empennage surfaces, and some control surfaces. In general, use of blind fasteners should keep to the absolute minimum. Panels which must be removable will require screw and nutplats; other applications with no back side accessability will require blind fasteners.

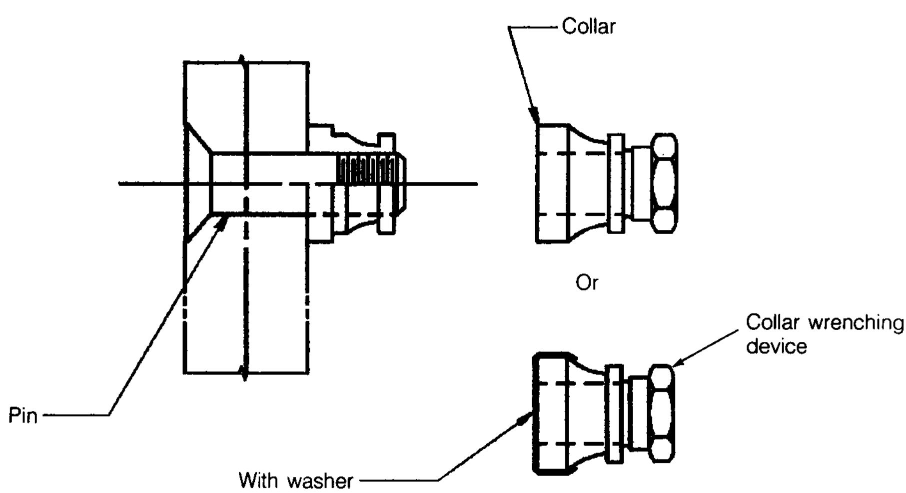
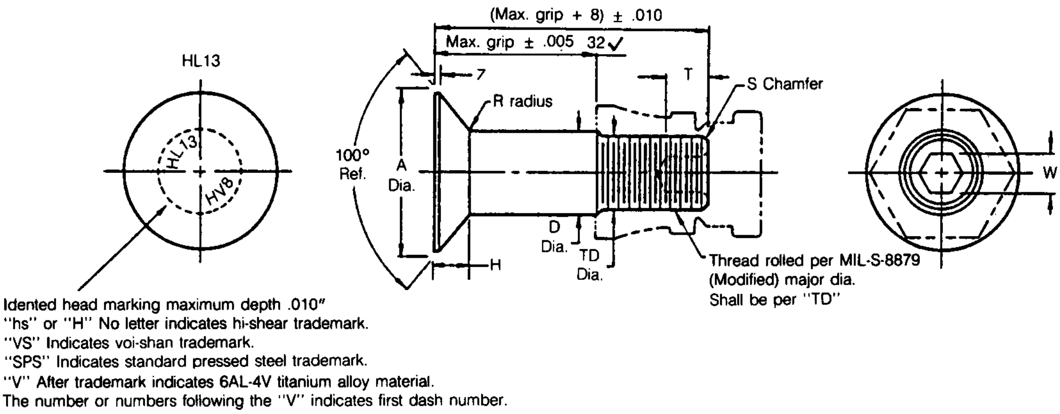


Fig. 5.2.13 HI-LOK Fastener



First dash No.	Nom. Dia.	A Dia.	B Ref.	D Dia. [8] Without solid film lube	D Dia. [8] With solid film lube	TD Dia.	F	H	R Rad.	Z	S Chamfer	Thread	SOCKET W Hex	SOCKET T Depth
−5	5/32	.3304 .3256	.312	.1635 .1630	.1635 .1625	.1595 .1570	.004	.0700 .0680	.025 .015	.010 .005	1/32"×45°	8-32UNJC-3A Modified	.0645 .0635	.135 .115
−6	3/16	.3813 .3765	.325	.1895 .1890	.1895 .1885	.1840 .1810	.005	.0805 .0785	.030 .020	.015 .005	1/32"×45°	10-32UNJF-3A Modified	.0806 .0791	.135 .115
−8	1/4	.5066 .5018	.395	.2495 .2490	.2495 .2485	.2440 .2410	.006	.1080 .1060	.030 .020	.015 .005	1/32"×45°	1/4-28UNJF-3A Modified	.0967 .0947	.150 .130
−10	5/16	.6335 .6287	.500	.3120 .3115	.3120 .3110	.3060 .3020	.007	.1350 .1330	.040 .030	.015 .005	3/64"×45°	5/16-24UNJE—3A Modified	.1295 .1270	.170 .150
−12	3/8	.7604 .7556	.545	.3745 .3740	.3745 .3735	.3680 .3640	.008	.1620 .1600	.040 .030	.015 .005	3/64"×45°	3/8-24UNJF-3A Modified	.1617 .1582	.200 .180
−14	7/16	.8884 .8812	.635	.4370 .4365	.4370 .4360	.4310 .4260	.009	.1895 .1865	.050 .040	.022 .005	3/64"×45°	7/16-20UNJF-3A Modified	.1930 .1895	.230 .210
−16	1/2	1.0139 1.0068	.685	.4995 .4990	.4995 .4985	.4830 .4880	.010	.2160 .2130	.050 .040	.022 .005	3/64"×45°	1/2-2UNJF-3A Modified	.2242 .2207	.260 .240
−18	9/16	1.1408 1.1337	.770	.5615 .5610	.5615 .5605	.5550 .5500	.010	.2430 .2400	.050 .040	.025 .005	1/18"×45°	9/16-18UNJF.3A Modified	.2555 .2520	.290 .270
−20	5/8	1.2723 1.2651	.825	.6240 .6230	.6240 .6230	.6180 .6120	.010	.2720 .2690	.050 .040	.025 .005	1/16"×45°	5/8-18UNJF-3A Modified	.2555 .2520	.330 .305

General notes:
1. Head edge out of roundnes shall not exceed "F".
2. Concentricity: conical surface of head to "D" diameter within .005 tir
3. "H" is dimensioned from maximum "D" diameter.
4. Dimensions of solid film lubed parts to be met after lube.
5. Surface texture per USASI B46.1.
6. Hole preparation per NAS618.
7. Use HL113 for oversize replacement.
8. Maximum "D" diameter may be increased by .0002 to allow for solid film application.

Material: 6AL-4V Titanium alloy per spec. AMS4928 or AMS4967.

Heat treat: 160,000 PSI tensile minimum (95,000 PSI shear minimum).

Finish: HL13V-( )-( ) — Cetyl alcohol lube per hi-shear spec. 305.

HL13VT-( )-( ) — Surface coating per hi-shear spec. 306, type I, color pink, and cetyl alchohol luber per hi-shear spec. 305.

(All diamensions are in inches)

Fig. 5.2.14 HI-LOK Fastener Data (HL13)

HL13VUE-( )-( )	—	Surface coating per hi-shear spec. 306, typell, and cetyl alcohol lube per hi-shear. 305.
HL13VV-( )-( )	—	Lubeco #2123 solid film lubicant.
HL13VR-( )-( )	—	Surface coating per hi-shear spec. 306, type II, and solid film lube per electrofilm, inc., spec. 4396
HL13VF-( )-( )	—	Surface coating per hi-shear spec. 306, type I, color blue, and cetyl alcohol lube per hi-shear spec. 305.

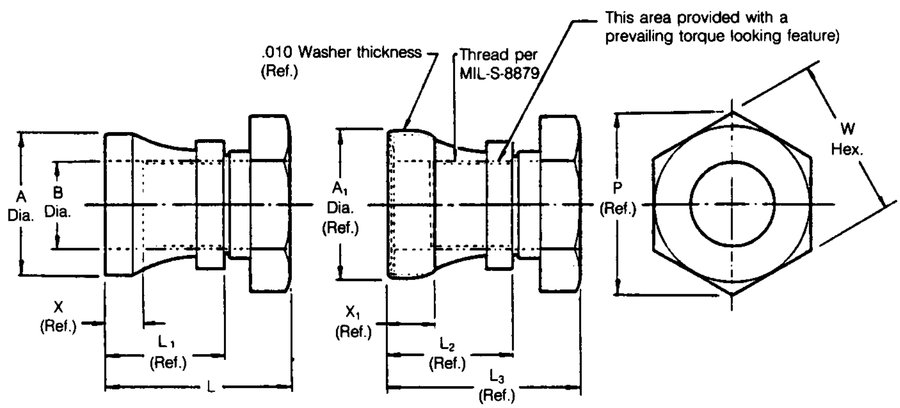
Specification: Hi-lok product spec. 340, section 4.

−5 Size must be installed using a torque controlled hex key.
See reference collar standards page for detail dimensions.

Code: First dash number indicates nominal diameter in 1/32NDS.
Second dash number indicates maximum in 1/16THS.
See "finish" note for explanation of code letters.

(Source: HI-Shear Corp.)

Fig. 5.2.14 HI-LOK Fastener Data (HL13) (cont'd)



Dash No.	Pin nom. Dia.	Thread	A Dia.	A_1 Dia. (Ref.)	B Dia.	L.	L_1 (Ref.)	L_2 (Ref.)	L_3 (Ref.)	P (Ref.)	W Hex.	X (Ref.)	X_1 (Ref.)
−5	5/32"	8-32UNJC-3B			note: for −5 diameter pin, use HL86-5.								
−6	13/64"	10-32UNJF-3B	.307 .303	.340	.212 .218	.457 .437	.275	.290	.470	3.44	.314 .302	.107	.120
−8	17/64"	1/4-28UNJF-3B	.412 .408	.442	.272 .268	.552 .532	.340	.355	.565	.380	.346 .332	.112	.125
−10	21/64"	5/16-24UNJF-3B	.518 .512	.552	.336 .330	.672 .652	.430	.445	.685	.484	.440 .425	.122	.135
−12	25/64"	3/8-24UNJF-3B	.628 .622	.665	.398 .392	.744 .724	.475	.490	.755	.557	.503 .488	.122	.135
−14	25/64"	7/16-20UNJF-3B	.733 .727	.775	.463 .457	.862 .842	.560	.575	.875	.840	.753 .736	.137	.150
−16	33/64"	1/2-20UNJF-3B	.848 .842	.895	.528 .522	.942 .922	.610	.625	.955	.970	.878 .861	.137	.150
−18	37/64"	9/16-18UNJF-3B	.878 .872	.910	.598 .592	1.029 1.009	.670	.685	1.042	.970	.878 .861	.145	.158
−20	41/64"	5/8-18UNJF-3B	1.005 .995	1.040	.661 .665	1.123 1.103	.735	.750	1.136	1.120	1.003 .986	.145	.158
−24	49/64"	3/4-16UNJF-3B	1.105 1.095	1.040	.786 .780	1.371 1.351	.940	.955	1.384	1.240	1.128 1.110	.156	.169
−28	57/64"	7/8-14UNJF-3B	1.295 1.285	1.330	.911 .905	1.571 1.551	1.060	1.075	1.584	1.450	1.315 1.292	.169	.182
−32	1-1/64"	1-12UNJF-3B	1.475 1.465	1.510	1.036 1.030	1.836 1.816	1.225	1.240	1.849	1.650	1.504 1.480	.186	.199

(All dimensions are in inches)

Fig. 5.2.15 HI-LOK Collar Data (HL87)

Notes: 1. Go thread gage penetration shall be 3/4 of one revolution minimum.
2. Dimensions apply after plating or solid film lube.

Material: Collar — 302 se stainless steel per QQ-S-763.
Washer — 302 stainless steel per MIL-S-5059.

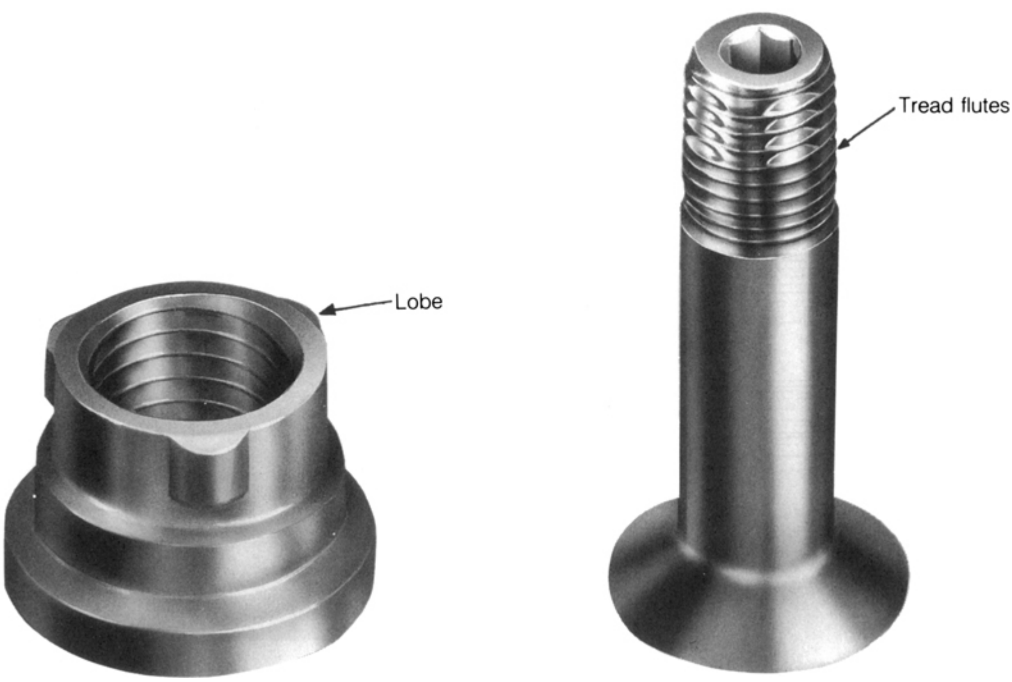
Finish: HL87-( ) — Collar only with cadmium plate per QQ-P-416, type II, class 3, and lauric acid or cetyl alcohol lube per hi-shear spec. 305.
HL87D-( ) — Collar only with cadmium plate per QQ-P-416, type II, class 3, and solid film lubricant per MIL-L-8937.
HL87DU-( ) — Collar only with solid film lube per MIL-L-8937.
HL87W-( ) — Collar with cadmium plate per QQ-P-416, type II, class 3, and lauric acid or cetyl alchohol lube per hi-shear spec. 305, and washer with cadmium plate per QQ-P-416, type I, class 3, no lube.

Specification: HI-LOK product spec. 340, section 101.
Code: Dash number indicates nominal thread size in 1/32NDS.
See "finish" note for explanation of code letters.

Example: HL87W-8 — 1/4-28 HI-LOK collar and washer.

(Source: HI SHEAR CORP.)

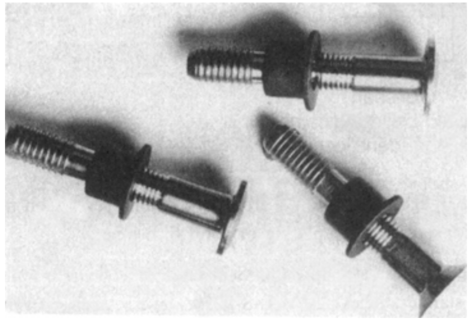
Fig. 5.2.15 HI-LOK Collar Data (HL87) (cont'd)

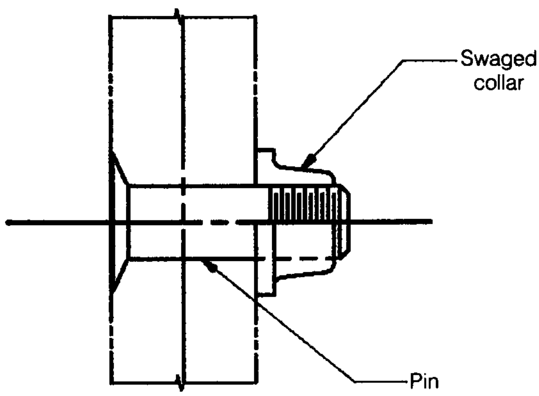


(a) Collar (b) Pin

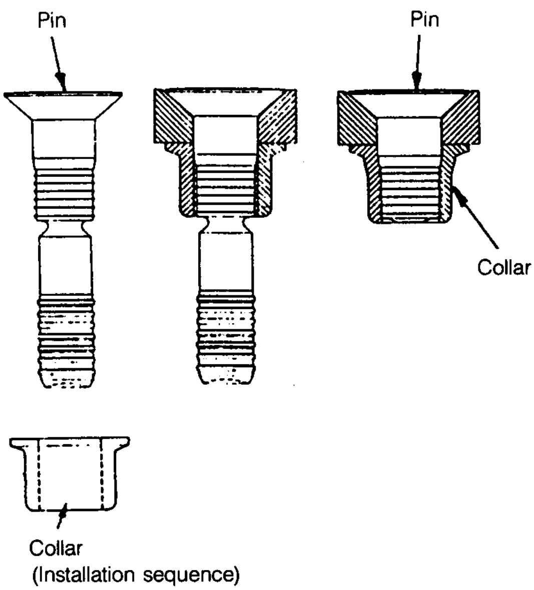
By courtesy of Fairchild Aerospace Fastener Division, Fairchild Corp.

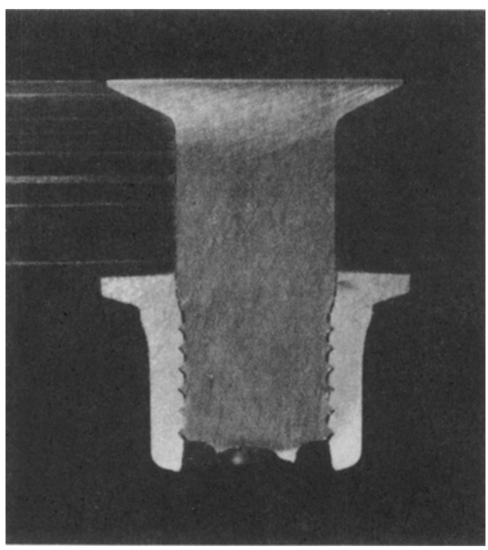
Fig. 5.2.16 Eddie-Bolt 2





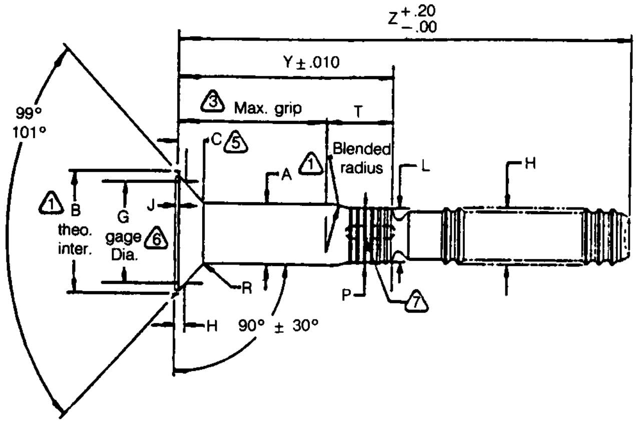
LGPL9SC (Clearance fit 0.001-0.003 inch)





By courtesy of HUCK Manufacturing Co.

Fig. 5.2.17 HUCK-COMP Fastener



LGP pin Family number	Nom. size	A Shank Dia.		B head Dia.	C head height	G Gage Dia.		H Gage height		J	L	M	P Dia.	R Radius		S 2	T
		Max.	Min.	Max.	Max.	Max.	Min.	Max.	Min.	Max.	Ref.	Max.	Max.	Max.	Min.		Ref.
LGPL8SC-V05B( )	.164	.1635	.1630	.332	.073	.2832	.2830	.0202	.0175	.010	.126	.156	.156	.025	.015	.0045	.175
LGPL8SC-V06V( )	.190	.1895	.1890	.383	.084	.3272	.3270	.0230	.0200	.013	.150	.184	.184	.030	.020	.0045	.176
LGPL8SC-V08B( )	.250	.2495	.2490	.510	.110	.4320	.4318	.0322	.0288	.017	.187	.244	.244	.030	.020	.0045	.244
LGPL8SC-V108( )	.312	.3120	.3115	.637	.137	.5451	.5449	.0378	.0342	.020	.244	.306	.306	.040	.030	.0045	.313
LGPL8SC-V12B( )	.375	.3745	.3740	.765	.165	.6582	.6580	.0439	.0401	.023	.298	.368	..370	.040	.030	.0060	.374

1 Concentricity: Conical surface of head to "A" diameter to be within .005 TIR.

2 Shank straightness: Within "S" values TIR per inch of shank length.

3 Grip length number: To determine the grip length number divide the dimensional thickness of parts being joined by .0625 inch. Grip length: To determine the grip length. Multiply the grip length number by .0625 inch. The grip length is measured from the top of the head to end the end of the full cylindrical portion of the shank.

4 See coding under usage and application for complete standard number.

5 Dimensions B & C for engineering reference only. Not for inspection purposes.

6 Drill center dimple permitted in head.

Material: 6AL-4V titanium alloy per AMS4967.
Minimum shear strength: 95 KSI.
Surface texture: Ra max. per ANSI B46. 1: Bearing surface, head to shank radius, and shank & lead-in radius −32; other surfaces −125.
Cetyl alcohol lube (chlorine free).
Head marking: Pin head shall be marked with HUCK'S basic symbol, basic number, and material designator, depressed .010 max. sealant escape groove: pin head shall be marked with HUCK'S basic symbol, basic number, material designator & H, depressed .010 max. arrangement optional. Example:

Example: LGP8 VB   7 LGP8 VBH

4 Usage and application information

Coding: The first set of letters designate the family of lightweight GP pins for shear or tension applications (LGPL).
The second set of letters & number 8SC designate head size, style, and load application.
The next letter, "V", is the material designator for 6AL-4V titanium alloy, min. Shear strength: 95 KSI.
The numbers following the material designator designates the nominal pin shank diameter in .0312 inch increments.
The final number(s) designates the grip length number or the nominal pin shank length in .0625 inch increments.

(All dimensions are in inches)

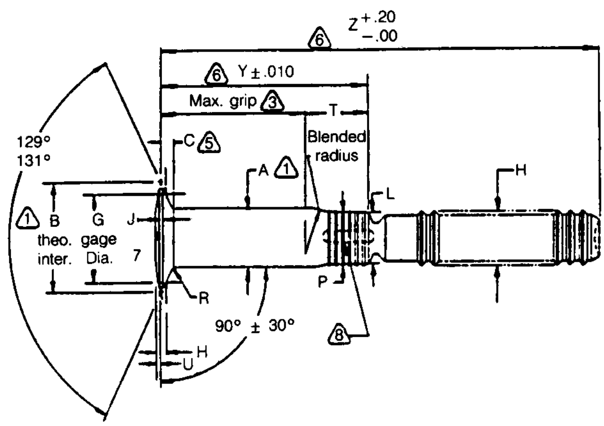
Fig. 5.2.18 HUCK-COMP Fastener Data (Tension Head)

Example: (H) LGPL 8SC V O8 B06

- .375 inch grip length.
- For composite applications.
- .250 inch shank diameter.
- 6AL-4V Titanium.
- Large 100° flush head.
- Family of LGP lightweight pins for aerospace applications.
- (7) The above is also available with sealant escape groove. Add prefix "H" to part number, example: HLGPL8SC-V08B06.

(Source: HUCK Manufacturing Co.)

Fig. 5.2.18 HUCK-COMP Fastener Data (Tension Head) (cont'd)



LGP pin Family number	Nom. size	A Shank Dia. Max.	A Shank Dia. Min.	B head Dia. Max.	C head height Max.	G Gage Dia. Max.	G Gage Dia. Min.	H Gage height Max.	H Gage height Min.	J Max.	L Ref.	M Max.	P Dia. Max.	R Radius Max.	R Radius Min.	S (2)	T Ref.	U Ref.
LGPL9SC-V05B( )	.164	.1635	.1630	.332	.040	.2560	.2550	.0263	.0235	.010	.126	.156	.156	.025	.015	.0045	.175	.006
LGPL9SC-V06V( )	.190	.1895	.1890	.383	.045	.2982	.2980	.0257	.2980	.013	.150	.184	.184	.030	.020	.0045	.176	.004
LGPL9SC-V08B( )	.250	.2495	.2490	.510	.110	.4320	.4318	.0322	.0288	.017	.187	.244	.244	.030	.020	.0045	.244	.005
LGPL9SC-V108( )	.312	.3120	.3115	.595	.066	.4791	.4789	.0350	.0308	.020	.244	.306	.306	.040	.030	.0045	.313	006
LGPL9SC-V12B( )	.375	.3745	.3740	.701	.076	.5942	.5940	.0343	.0293	.023	.298	.368	.370	.040	.030	.0060	.374	.007

(1) Concentricity: Conical surface of head to "A" diameter to be within .005 TIR.

(2) Shank straightness: Within "S" values TIR per inch of shank length.

(3) Grip length number: To determine the grip length number divide the dimensional thickness of parts being joined by .0625 inch. Grip length: To determine the grip length. Multiply the grip length number by .0625 inch. The grip length is measured from the theoretical intersection of crown & head angle to the end the of the full cylindrical portion of the shank.

(4) See coding under usage and application for complete standard number.

(5) Dimensions B & C for engineering reference only. Not for inspection purposes.

(6) "Y" & "Z" are located from theoretical intersection of crown & head angle.

(7) Drill center dimple permitted in head.

(All dimensions are in inches)

Fig. 5.2.19 HUCK-COMP Fastener Data

Material: 6AL-4V titanium alloy per AMS4967.

Minimum shear strength: 95 KSI.

Surface texture: Ra max. per ANSI B46. 1: Bearing surface, head to shank radius, and shank & lead-in radius −32; other surfaces −125.

Cetyl alcohol lube (chlorine free).

Head marking: Pin head shall be marked with HUCK's basic symbol, basic number, and material designator, depressed .010 max. sealant escape groove: pin head shall be marked with HUCK'S basic symbol, basic number, material designator & H, depressed .010 max. arrangement optional. Example:

Example: (LGP9 VB) 8 (LGP9 VBH)

4 Usage and application information

Coding: The first set of letters designate the family of lightweight GP pins for shear or tension applications (LGPL).
The second set of letters & number 9SC designate head size, style, and load application.
The next letter, "V", is the material designator for 6AL-4V titanium alloy, min. Shear strength: 95 KSI.
The numbers following the material designator designates the nominal pin shank diameter in .0312 inch increments.
The final number(s) designates the grip length number or the nominal pin shank length in .0625 inch increments.

Example: (H) LGPL 9SC V O8 B06

- B06 — .375 inch grip length.
- B — For composite applicataions.
- 08 — .250 inch shank diameter.
- V — 6AL-4V Titanium.
- 9SC — Large profile 130° flush head, for shear applications.
- LGPL — Family of LGP lightweight pins for aerospace applications.
- (H) 8 — The above is also available with sealant escape groove. Add prefix "H" to part number, example: HLGPL9SC-V08B06.

(Source: HUCK Manufacturing Co.)

Fig. 5.2.19 HUCK-COMP Fastener Data (cont'd)

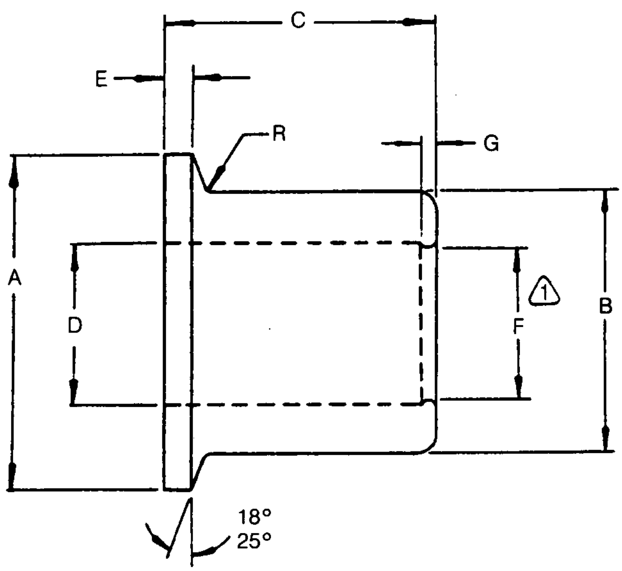


Fig. 5.2.20 Collar Data for Both HUCK-COMP and HUCK-TITE Fasteners

Identification: Manufacturing option to identify collar with X.
△1 Permissible punchout burr permitted within limits of "F" & "G".

Part number	Nominal size	A	B	C	D	E	F	G	R	Install with
SLFC-MV05	.164	.334 .314	.248 .244	.235 .215	.1625 .1655	.020 .030	.159	.016	.008 .018	LGP( )8S( )-V05B LGP( )9S( )-V05B
SLFC-MV06	.190	.388 .368	.274 .270	.240 .220	.1880 .1910	.020 .030	.186	.016	.008 .018	LGP( )8S( )-V06B LGP( )9S( )-V06B
SLFC-MV08	.250	.477 .457	.357 .353	.300 .280	.2485 .2515	.025 .035	.246	.016	.008 .018	LGP( )8S( )-V08B LGP( )9S( )-V08B
SLFC-MV10	.312	.604 .584	.448 .444	.370 .350	.3100 .3130	.025 .035	.308	.031	.013 .023	LGP( )8S( )-V10B LGP( )9S( )-V10B
SLFC-MV12	.375	.701 .681	.533 .529	.420 .400	.3725 .3755	.030 .040	.370	.031	.013 .023	LGP( )8S( )-V12B LGP( )9S( )-V12B

Material: Cp titanium
Finish: None
Lubrication: Dry film per MIL-L-8937 plus chlorine free cetyl alcohol.
Surface texture: AA max. per ANSI B46.1-125,
Suggested temperature limitations: Room temperature to 250°F.

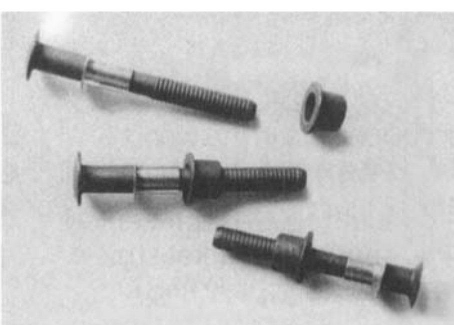
Shear flanged Collar part no. code (and example)

SLFC-MV06
- 06 — Nominal pin shank in 32nd inches.
- MV — Material: cp titanium
- SLFC — Shear flanged collar.

Quality control requirements as applicable per HUCK quality control manual, established by the requirements of specification MIL-Q-9858 and other governing specifications.

(Source: HUCK Manufacturing Co.)

Fig. 5.2.20 Collar Data for both HUCK-COMP and HUCK-TITE Fasteners (cont'd)



HUCK LGPL Lockbolt
with expandable sleeve
(0.001-0.005 inch interference)

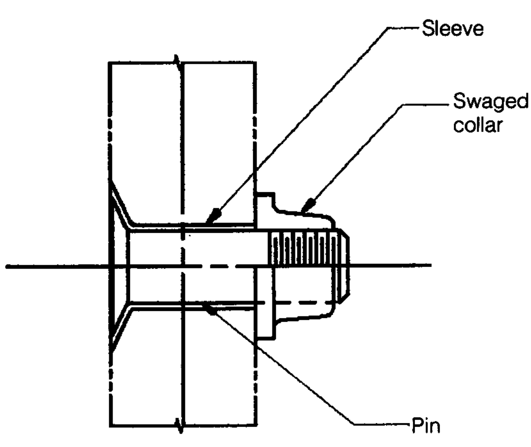
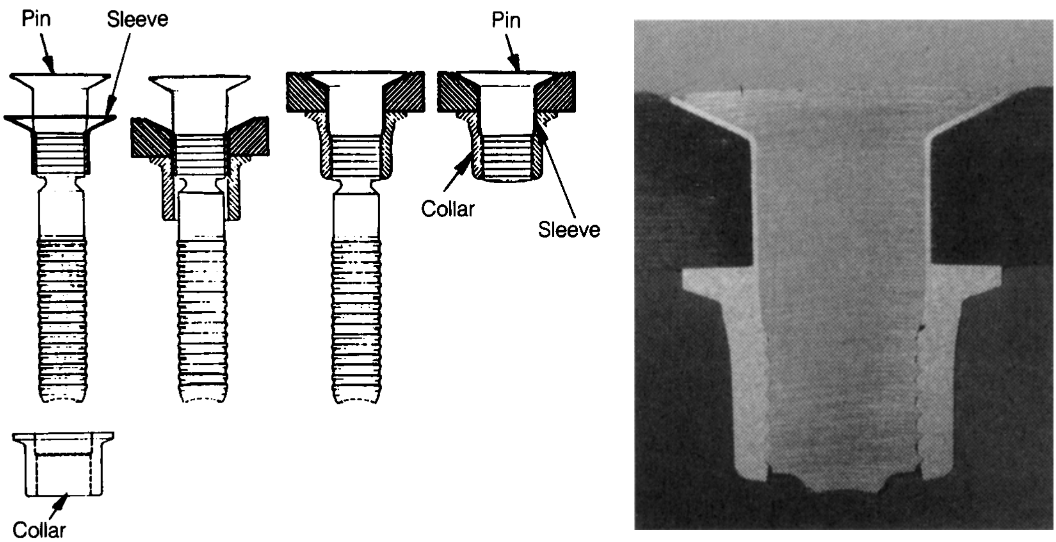


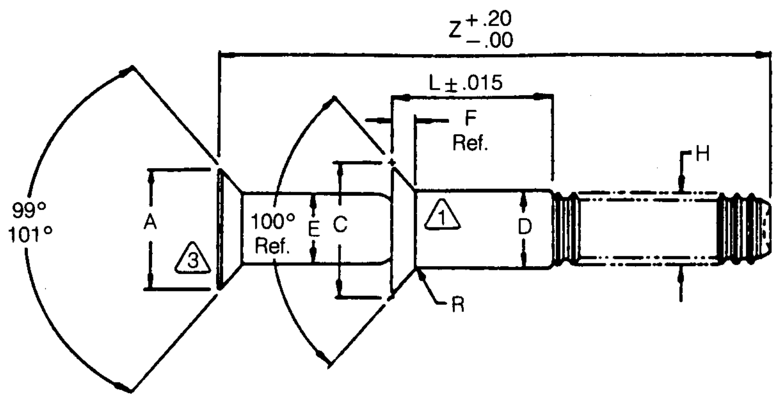
Fig. 5.2.21 HUCK-TITE Interference Fit Fastener



(Installation sequence)

By courtesy of HUCK Manufacturing Co.

Fig. 5.2.21 HUCK-TITE Interference Fit Fastener (cont'd)



CIL fastener part number	A	C Theo. int. Ref.	D	E Min.	F Ref.	M Max.	A Radus	
							Max.	Min
CIL8SC-VC05-( )	.317 .307	.353	.1755 .1745	.1650	.075	.156	.022	.012
CIL8SC-VC06-( )	.367 .357	.403	.2015 .2005	.1910	.085	.184	.027	.017
CIL8SC-VC08-( )	.486 .476	.529	.2625 .2615	.2518	.112	.244	.027	.017
CIL8SC-VC10-( )	.605 .595	.656	.3235 .3225	.3128	.140	.306	.037	.027
CIL8SC-VC12-( )	.725 .715	.783	.3875 .3865	.3767	.166	.368	.037	.027

Fig. 5.2.22 HUCK-TITE Fastener Data (Interference Fit Flush Head)

1. Sleeve O.D. may be distorted due to crimping the sleeve to the pin.
2. See coding under usage and application for complete standard number.
3. Drill center dimple permitted in head.
4. Use with CIFC-MV( ) collar. see SK12296-4 for installation procedure.

Material pin: 6AL-4V titanium alloy per AMS4967.
Minimum shear strength pin: 95 KSI.
Finish pin: drifilm per MIL-L-8937 plus cetyl alcohol.

Sleeve: A286 CRES (A.I.S.I. 660) per AMS5525.
Sleeve: Full anneal.
Sleeve: Bare. passivate per QQ-P-35.

Surface finish: Ra max. per ANSI B46.1: bearing surface, head to shank radius, and shank −32; other surfaces — 125.
Head marking: Fastener head shall be marked with HUCK'S basic symbol, basic number and material designator, depressed .010 max.

Example: CIL8 V

2 Usage and application information

Coding: The first set of letters designate the family of composite interference fit fasteners for shear or tension applications (CIL)
The second set of letters & numbers (8SC) designate head size, style and load application.
The next letters "VC" are the material designators for 6AL-4V titanum alloy pin and A-286 CRES sleeve.
The numbers following the material designator designate the shank diameter.
The final number(s) designates the grip length number in .0625 inch increments.

Example: CIL 8SC-VC 08-06

- .375 inch grip length.
- .262 inch shank diameter.
- 6AL-4V titanium pinia-286 CRES sleeve.
- Large 100° flush head for shear applications.
- Family of composite interference fit fasteners.

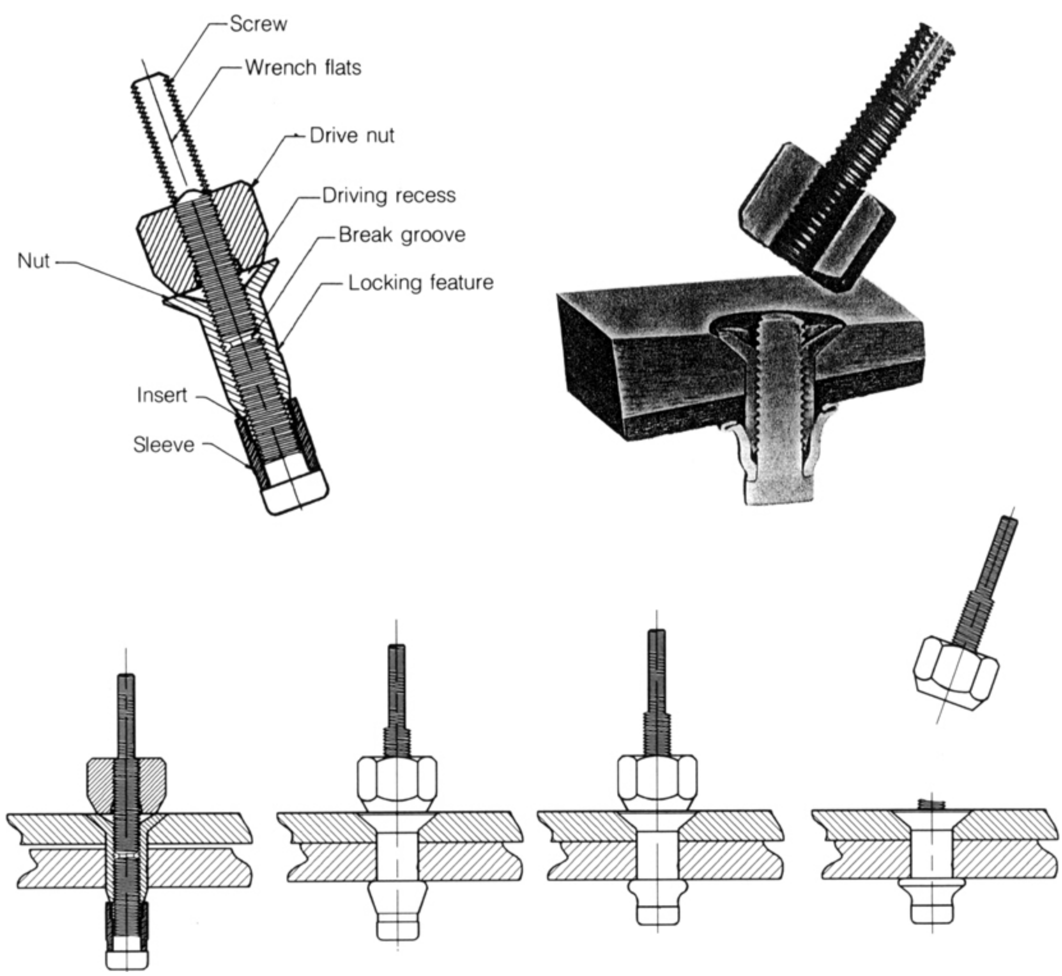
(Source: HUCK Manufacturing Co.)

Fig. 5.2.22 HUCK-TITE Fastener Data (Interference Fit Flush Head) (cont'd)

The conventional blind fastener consisting of a tubular sleeve in which a stem having an enlarged end cannot be used for composite structural assembly due to insufficient blind head expansion. In addition, they may damage the composite laminates during installation. This damage is caused by either excessive clamp load for the available bearing area or by radial fastener expansion within the fastener hole, which casues the composite plies to delaminated on the blind-side surface.

Fig. 5.2.23 and 5.2.24 show a COMPOSI-LOK II blind fastener specifically designed to have a large, blind side footprint for application in carbon composite structures. It has a stainless stem and the back-side clamping sleeve folds under when it hits the laminate surface to provide an axial-flow bulbing action which is enhanced semi-hydraulically by a trapped internal plastic sleelve.

Fig. 5.2.25 shows a screw-stem COMP-TITE blind fastener system designed specifically for composites with a titanium threaded sleeve and stainless stem, back side sleeve and washer. The blind side footprint is formed by expansion of a flat-faced helical washer backed up by an expansion sleeve. This washer forms a large bearing surface that will not crush or delaminate composites during installation.



By courtesy of Monogram Aerospace Fasteners

Fig. 5.2.23 COMPOSI-LOK II Big Foot Blind Fastener

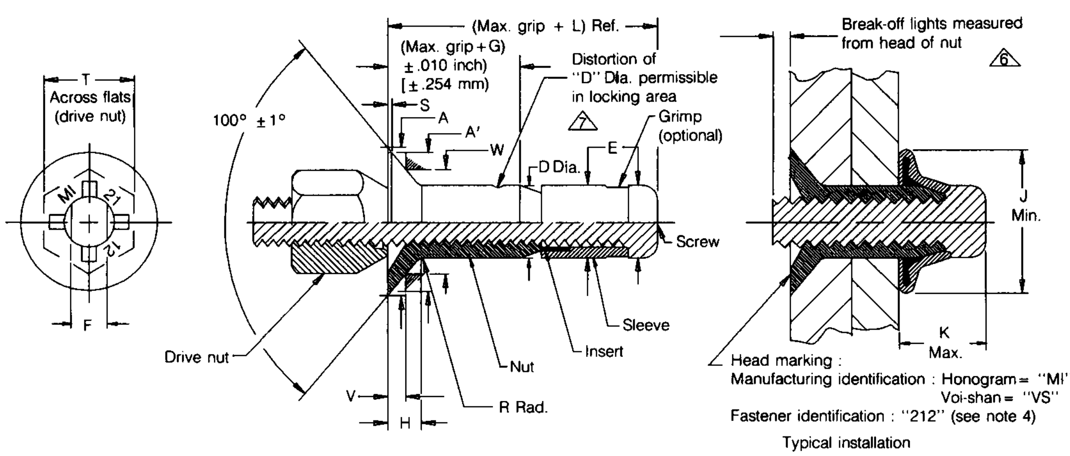


Fig. 5.2.24 COMPOSI-LOK II Blind Fastener Data

TABLE I

PART NUMBER	A DIA. THEO. INCH	A DIA. THEO. mm	A' DIA. MIN. INCH	A' DIA. MIN. mm	D DIA. INCH	D DIA. mm	E DIA. MAX. INCH	E DIA. MAX. mm	F WRENCH FLATS INCH	F WRENCH FLATS mm	G INCH	G mm	H REF. INCH	H REF. mm	L INCH	L mm	R RAD. MAX. INCH	R RAD. MAX. mm	S MAX. INCH	S MAX. mm	T ACROSS HEX. INCH	T ACROSS HEX. mm
MBF2112-5-()	.332 .325	8,43 8,25	.296	7,52	.1645 .1625	4,178 4,128	.1640	4,166	.085 .080	2,16 2,03	.017	0,43	.070	1,78	.512	13,00	.030	0,76	.015	0,38	.375	9,52
MBF2112-6-()	.385 .378	9,78 9,60	.342	8,69	.1985 .1965	5,042 4,991	.1985	5,042	.113 .108	2,87 2,74	.027	0,68	.077	1,96	.575	14,61	.030	0,76	.019	0,48	.375	9,52
MBF2112-7-()	.416 .409	10,57 10,39	.373	9,47	.2275 .2255	5,778 5,728	.2275	5,778	.121 .116	3,07 2,95	.035	0,89	.077	1,96	.635	16,13	.030	0,76	.020	0,51	.375	9,52
MBF2112-8-()	.507 .499	12,88 12,67	.463	11,76	.2595 .2575	6,591 6,541	.2595	6,591	.135 .130	3,43 3,30	.055	1,40	.104	2,64	.700	17,78	.030	0,76	.020	0,51	.375	9,52
MBF2112-9-()	.538 .530	13,66 13,46	.494	12,55	.2895 .2875	7,353 7,303	.2895	7,353	.152 .147	3,86 3,73	.065	1,65	.104	2,64	.815	20,70	.030	0,76	.020	0,51	.500	12,70
MBF2112-10-()	.635 .626	16,13 15,90	.577	14,66	.3115 .3095	7,912 7,861	.3110	7,899	.152 .147	3,86 3,73	.070	1,78	.136	3,45	.892	22,66	.040	1,02	.026	0,66	.500	12,70
MBF2112-11-()	.666 .657	16,92 16,69	.608	15,44	.3435 .3415	8,725 8,674	.3433	8,720	.185 .180	4,70 4,57	.075	1,90	.136	3,45	.941	23,90	.040	1,02	.026	0,66	.500	12,70
MBF2112-12-()	.762 .752	19,35 19,10	.696	17,68	.3745 .3725	9,512 9,462	.3740	9,500	.185 .180	4,70 4,57	.080	2,03	.162	4,11	1.090	27,69	.040	1,02	.029	0,74	.500	12,70

TABLE I (CONT)

PART NUMBER	MINIMUM AVAILABLE GRIP DASH NO.	INSTALLED DIMENSIONS: RECOMMENDED HOLE SIZE INCH	RECOMMENDED HOLE SIZE mm	J DIA. MIN. INCH	J DIA. MIN. mm	K MAX. INCH	K MAX. mm	BREAK-OFF LIMITS 6 INCH	BREAK-OFF LIMITS 6 mm	MECHANICAL PROPERTIES: TENSILE STRUCTURAL FAILURE (MIN.) LBS.	TENSILE STRUCTURAL FAILURE (MIN.) N	DOUBLE SHEAR MIN. LBS.	DOUBLE SHEAR MIN. N	LOCKING TORQUE MIN. IN-LBS	LOCKING TORQUE MIN. Nm	V GAGE PROT. INCH	V GAGE PROT. mm	W GAGE DIA. INCH	W GAGE DIA. mm
MBF2112-5-()	-150	.168 .165	4,27 4,19	.250	6,35	.300	7,62	+.103 -.000	+2,62 -0,00	900	4000	3150	14010	1.0	0,113	.0207 .0174	0,526 0,442	.2832 .2830	7,193 7,188
MBF2112-6-()	-150	.202 .199	5,13 5,05	.300	7,62	.350	8,89	+.103 -.000	+2,62 -0,00	1400	6230	4600	20460	1.5	0,170	.0245 .0212	0,622 0,538	.3272 .3270	8,311 8,306
MBF2112-7-()	-150	.231 .228	5,88 5,79	.350	8,89	.400	10,16	+.103 -.000	+2,62 -0,00	1600	7120	6050	26910	2.0	0,226	.0358 .0324	0,909 0,823	.3315 .3313	8,420 8,415
MBF2112-8-()	-200	.263 .260	6,68 6,60	.400	10,16	.450	11,43	+.103 -.000	+2,62 -0,00	2100	9340	7900	35140	2.5	0,282	.0318 .0279	0,808 0,709	.4320 .4318	10,973 10,968
MBF2112-9-()	-200	.293 .290	7,44 7,37	.450	11,43	.500	12,70	+.103 -.000	+2,62 -0,00	2600	11565	9800	43590	3.0	0,339	.0446 .0407	1,133 1,034	.4320 .4318	10,973 10,968
MBF2112-10-()	-250	.315 .312	8,00 7,92	.475	12,06	.550	13,97	+.103 -.000	+2,62 -0,00	3600	16010	11350	50480	3.5	0,400	.0405 .0365	1,029 0,927	.5389 .5385	13,688 13,678
MBF2112-11-()	-250	.347 .344	8,81 8,74	.525	13,33	.575	14,60	+.103 -.000	+2,62 -0,00	4400	19570	13850	61600	4.0	0,452	.0539 .0500	1,369 1,270	.5389 .5385	13,688 13,678
MBF2112-12-()	-250	.378 .375	9,60 9,52	.575	14,60	.625	15,87	+.103 -.000	+2,62 -0,00	5000	22240	16450	73170	4.0	0,452	.0458 .0415	1,163 1,054	.6532 .6528	16,591 16,581

TABLE II

2ND DASH NO. (GRIP)	GRIP RANGE: MIN. GRIP INCH	MIN. GRIP mm	MAX. GRIP INCH	MAX. GRIP mm
100	.050	1,27	.100	2,54
150	.100	2,54	.150	3,81
200	.150	3,81	.200	5,08
250	.200	5,08	.250	6,35
300	.250	6,35	.300	7,62
350	.300	7,62	.350	8,89
400	.350	8,89	.400	10,16
450	.400	10,16	.450	11,43
500	.450	11,43	.500	12,70
550	.500	12,70	.550	13,97
600	.550	13,97	.600	15,24
650	.600	15,24	.650	16,51
700	.650	16,51	.700	17,78
750	.700	17,78	.750	19,05
800	.750	19,05	.800	20,32
850	.800	20,32	.850	21,59
900	.850	21,59	.900	22,86
950	.900	22,86	.950	24,13
1000	.950	24,13	1.000	25,40
1050	1.000	25,40	1.050	26,67
1100	1.050	26,67	1.100	27,94
1150	1.100	27,94	1.150	29,21

Procurement specification: MBF 2000

Installation & inspection specification: MBF 2001

Material and heat treat: .... Nut: 6AL-4V titanium per MIL-T-9047, STA, or AMS 4928 or AMS 4957. Heat treated per MIL-H-81299. Maximum hydrogen 125 PPM.

Screw: A-286 per AMS 5732, AMS 5731 or AMS 5737 heat treated to 175 KSI tensile minimum.

Sleeve: 304 stainless steel per AMS 5639. Fully annealed.

Insert: Acetal per ASTMD 4181.

Drive Nut: Mild steel (colorgray)

Finish: .... Nut: Kal-gard ANN-RQ #1012 conversion coating or phosphate fluoride per boeing specification PS 741 may be used at manufacturer's option.

Sleeve & Screw: Passive per QQ-P-35, Kal-gard conversion coating Kao-gard ANN-RO #1013 optional.

Insert: None.

Drive Nut: Colorgray

Fig. 5.2.24 COMPOSI-LOK II Blind Fastener Data (cont'd)

Lubricants: Tio-lube 460, dry film lube per Mil-L-8937 paraffin wax, cetyl alcohol, used as required for performance.

General Notes: 1.) Example of part number:

MBF 2112 S — 6 — 100

- 100 — Designates grip (table II)
- — — See note /6\
- 6 — Designates basic dimensions (table I)
- S — Designates special finish code (see HBF 2002)
- MBF 2112 — Designates basic part number

2.) Locking feature consists of three (3) indentations located 120° apart on the periphery of the nut component and approximately .040 above the intersection of the nut nose angle and O.D.

3.) See MBF 2003 for installation and removal information.

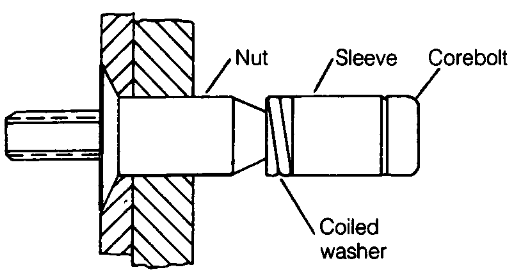
4.) Alternate head marking: "MI" (or "VS") and "2012".

5.) COMPOSI-LOK fasteners with selected combinations of the above lubricants and finishes are specially coded and may be substituted for equivalent non-coded parts at manufacturer's option. see interchangeability specification MBF 2007.

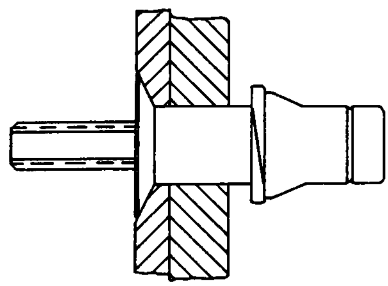
/6\ An "L" in place of the dash (—) between the diameter dash number and the grip dash number designates modified break-off limits of +.053/−.050. "e.g. MBF 2112 ( )-6L200".

/7\ Distortion shall not prevent insertion of the fastener into a ring gauge of length equal to max. grip and diameter equal to minimum recommended hole.
Force for insertion shall not exceed 5.0 pounds.

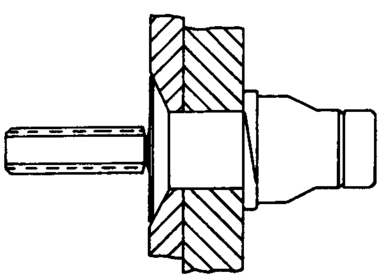
Fig. 5.2.24 COMPOSI-LOK II Blind Fastener Data (cont'd)



(1) Using NAS 1675-type installation tooling, the nuts is restrained from turning while the corebolt is driven.



(2) The advance of the croebolt forces the washer and sleeve over the taper, expanding and uncoiling the washer to its maximum diameter.



(3) Continued advanced of the corebolt draws the washer and sleeve against the joint surface, preloading the structure.

At a torque level controlled by the break groove, the slabbed portion of the corebolt separates, and installation is complete.

(blind fastener installation sequence.)

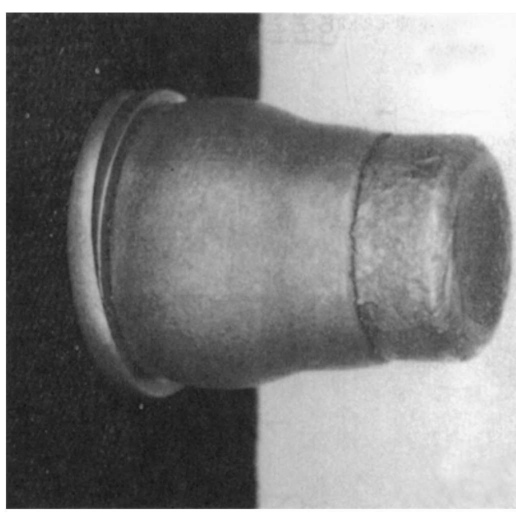
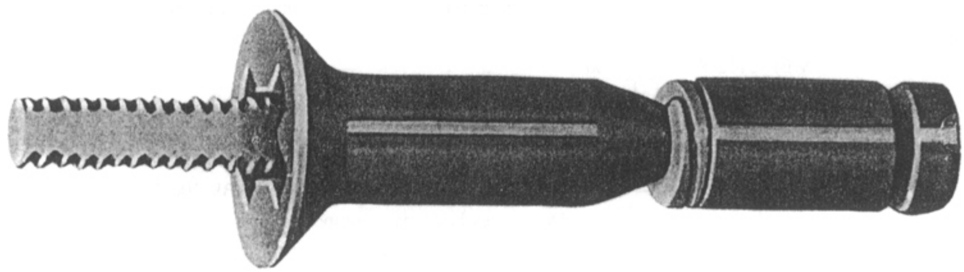


Fig. 5.2.25 COMP-TITE Blind Fastener



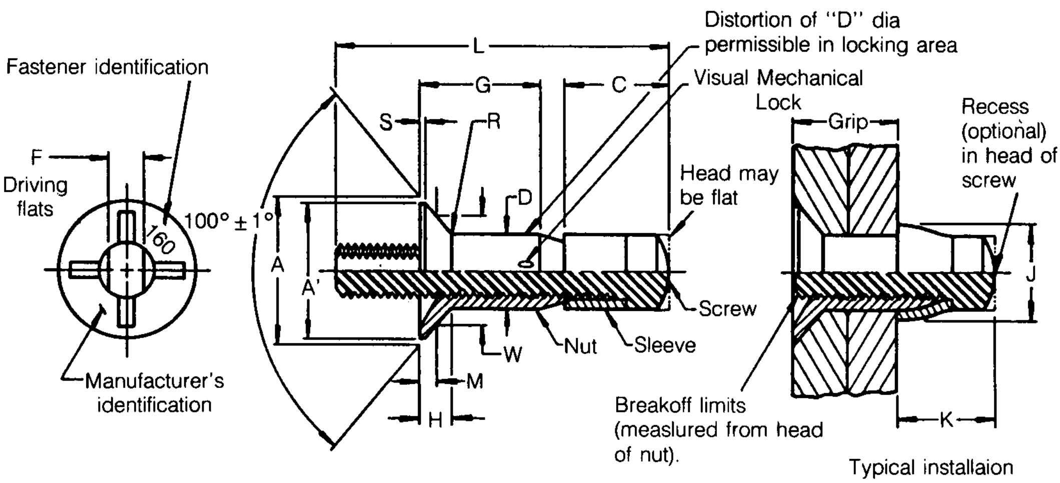
By courtesy of SPS Technologies Aerospace products Division

Fig. 5.2.25 COMP-TITE Blind Fastener (cont'd)

Fig. 5.2.26 shows a VISU-LOK blind fastener which is a version of the screw-stem bolt with a separate back side clamping sleeve. It consists of a titanium basic sleeve with a titanium stem and stainless back-side clamping sleeve.

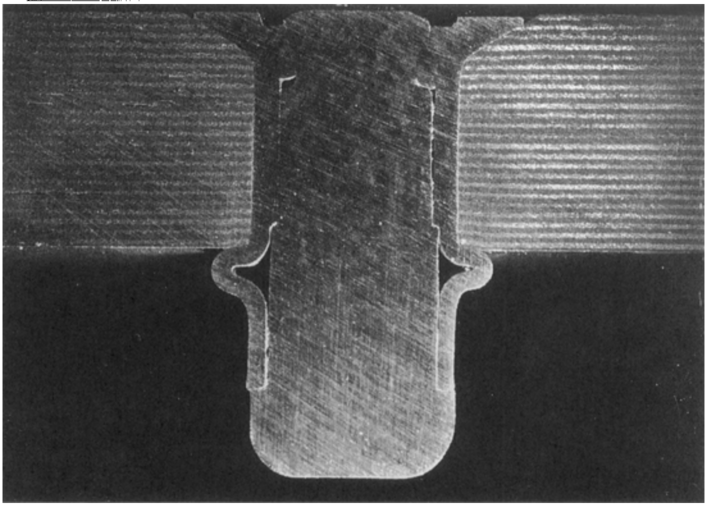
Fig. 5.2.27 shows a COMPOSI-BOLT blind bolt which is a blind fastener with a pull-stem, single-action and single sleeve bulbing action on the blind side. It is designed specifically for composite structures. Large bearing area available in the 100° and 120° shear and tension head are suitable for composite applications. The HUCK-TIMATIC blind bolt is a titanium version of the single action UNIMATIC blind bolt. It is available with a CP titanium sleeve and an A286 pin or a 15-3-3-3 titanium pin which was designed specifically for use in composite structures.

Only a few fasteners are available to be used for composite structural assembly and this Section provides basic design data for several of the commonly used fastener systems. However, the data given in Fig. 5.2.13 through 5.2.27 are vendor information and should only be used with caution.



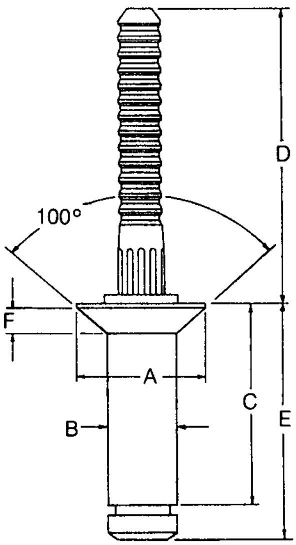
By courtesy of Fairchild Aerospace Fastener Division, Fairchild Corp.

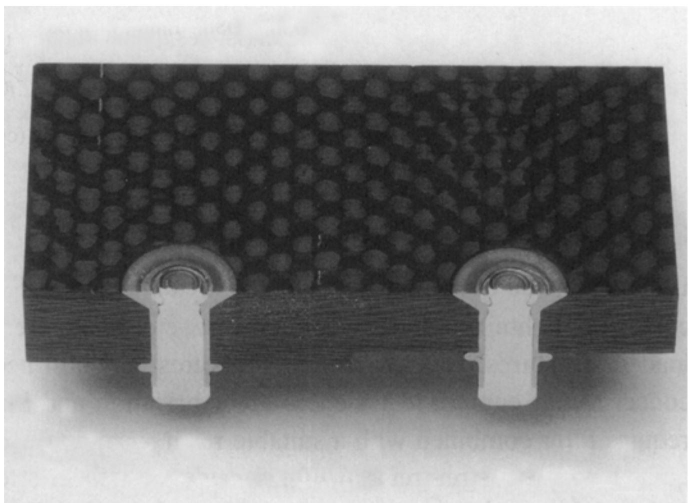
Fig. 5.2.26 VISU-LOK Blind Fastener



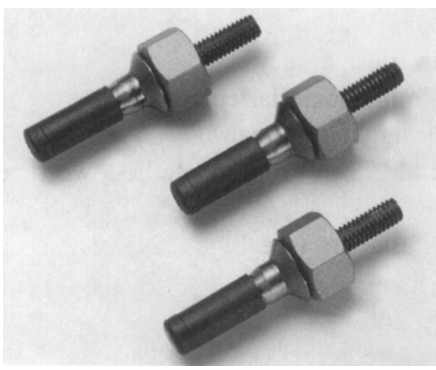
(Installed in Graphite Composite)

(a) UNIMATIC Blind Bolt





(b) Ti-matic Blind Bolt



(c) COMPOSI-BOLT Blind Bolt

Fig. 5.2.27 HUCK Blind Fastener

Chapter 6.0

ENVIRONMENTS

6.1 INTRODUCTION

Composites for airframe structures must be designed to withstand the great diversity of terrestrial environments encountered in a variety of operations. Environmental effects, including combinations of heat, cold, moisture, lightning strikes, ultraviolet (UV) radiation, fluids and fuels, can reduce mechanical properties to varying degrees, depending on the composite system.

A typical environmental effect is the hot/wet condition in which the elasticity and strength design allowables of the composite may reduce as much as 10% to 20% (50% reduction occur in some wet-layup materials). The composite matrix generally is the component most vulerable to environmental effects. Consequently matrix dominated properties such as loss of compressive strength are of greatest concern.

Another environmental concern is lightning strike (which occur once per year on the average for commercial aircraft; less frequently for military aircraft) — catastrophic failure of the aircraft structures can result.

The most common environmental hazards are listed below:

- Exposure to moisture
- Elevated temperatures
- Cold temperatures
- Lightning strikes and Electromagnetic interference (EMI)
- Sunlight — ultraviolet radiation
- Erosion of material
- Solvent and chemicals
- Ballistic impact
- Acoustic exposure
- Nuclear exposure

This chapter describes the main environmental hazards and problems peculiar to composite airframe structures, and recommends protective measures.

6.2 WEATHERING EFFECTS

In the terrestrial environment the combined effects of temperature and humidity must be considered when assessing long-term structural integrity.

Moisture/Temperature

The amount of moisture that composite materials absorb is a function of matrix and fiber type, time, component geometry, temperature, relative humidity, and exposure conditions. The absorption of moisture by the organic matrix of composites is an important factor at high temperatures. The moisture diffuses into the matrix, in both laminates and adhesives, causing them to swell and acts as a plasticizer or softener. The latter phenomenon is by far the most important for airframe applications and results in a decrease of the matrix glass transition temperature (Tg). This manifests itself in a decrease of matrix-dominated properties at high temperatures. For example, compression strength is clearly reduced, but tension behavior is relatively unaffected, at the upper temperature ranges of the material's usefulness.

Composite mechanical properties are dependent upon the amount of water (moisture) in the matrix and the temperature. The phenomenon may include some or all of the following:

- Composite matrix absorbs moisture until an equilibrium (or saturation) point is attained (see Fig. 6.2.1)
- Moisture lowers the glass transition temperature (Tg)
- Generally,
  — 350°F composites absorb more moisture than 250°F composites
  — Thermosets absorb 1% to 2% moisture
  — Thermoplastics absorb 0.1% to 0.3% moisture
- Hot/wet conditions cause the matrix to become more plastic
- Cold/dry conditions cause the matrix to become more brittle
- Moisture desorption gradients may induce microcracking as the surface desorbs and shrinks, putting the surface in tension. If the residual tension stress at the surface is beyond the strength of the matrix, cracks occur
- Cyclical swelling and contracting due to recurring moisture exposure may lead to joint loosening (a problem which is more critical with thermosets than thermoplastics)
- Lower design allowables or a protection system are required for composites which will be exposed to, if temperatures higher than normal design service temperatures since the material mechanical properties degrade after a very short time.
- To minimize moisture ingress:
  — Seal machined edges and surfaces of laminates
  — Provide surface paint or protective coating. However, paint will not prevent moisture from diffusing into the composite matrix

Some composite laminates may need to be protected in order to be resistant to fluids such as moisture, fuel, hydraulic fluid and anti-icing fluid. The following precautions should be taken for composite laminates:

- When a part is not painted, a thin film of thermoset or thermoplastic resin should be applied to the surface to reduce porosity (preventing ingress of moisture through surface cracks)
- Sealant should be applied when surfaces are exposed to weathering to reduce the effects of static rain, air oxidation, airblown sand, and ultraviolet radiation

LIVE GRAPH
Click here to view

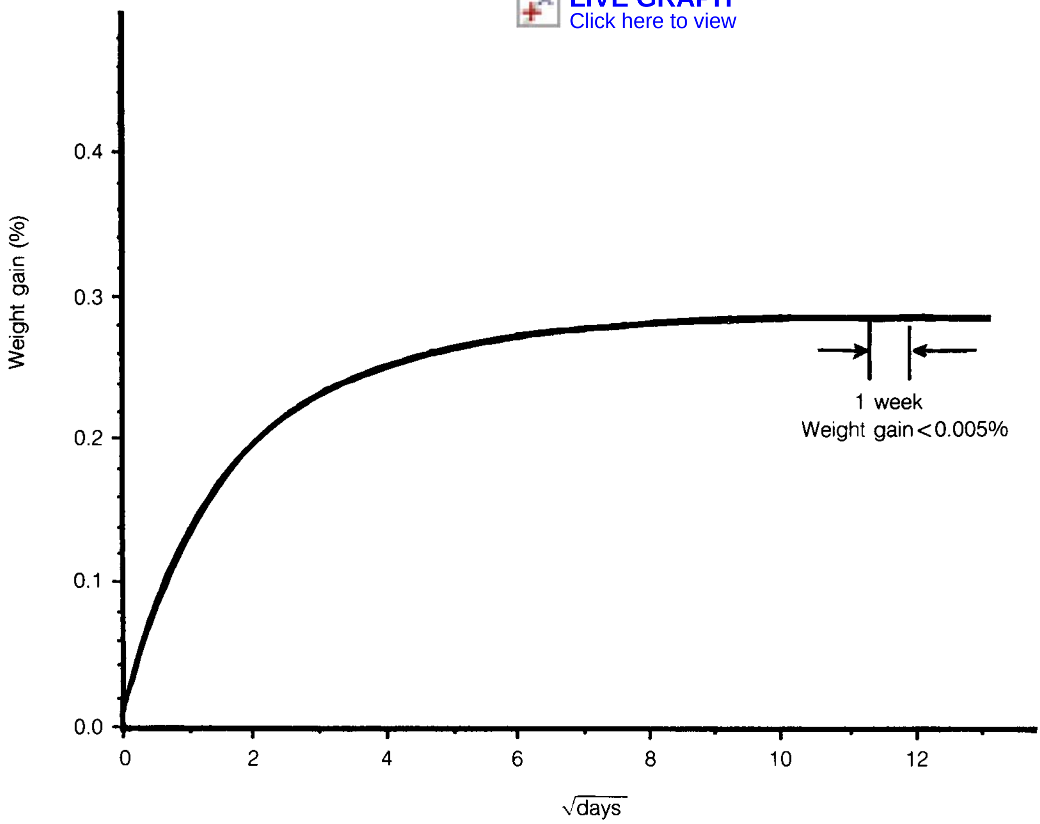


Fig. 6.2.1 Typical Equilibrium Moisture Content of a Thermoplastic

Ultraviolet Radiation

- Ultraviolet (UV) can deteriorate composite material integrity
- Carbon, glass and boron are impervious to UV, but Kevlar 49 is degraded by UV
- Presence of temperature/moisture with UV can magnifies degradation of material
- Conventional aircraft paint provides protection against UV

Hail and Foreign Objects Damage

The leading edge components of wings and empennage are vulnerable to rain, hail and foreign object damage. Complete impact protection is difficult to achieve and the equivalent thickness of composite part should be equal or near to of aluminium counterpart [e.g., 0.06 inch (1.52 mm) for large transport and will be less for small aircraft]. Fig. 6.2.2 shows the area on the leading edge (between points A and B) that is most susceptible to rain or hail damage.

Occasionally control surfaces such as ailerons, elevators, and rudders shows evidence of hailstorm including actual damage, puncture of the skin. The need for protection of control surfaces should not be overlooked.

Rain and Sand Erosion

Erosion of composites edge results from exposure to rain, dust, and sand (in desert areas). Proventative measures for structures which will experience long term exposure are given below:

- Aircraft leading edge components should be protected by rubberized coatings (see Fig. 6.2.3)
- Composite surfaces such as leading edges of wing, horizontal or vertical stabilizers, etc. require some form of protection when exposed to the airstream. Wherever possible, the best protection for leading edge composites is a metallic outer layer (see Fig. 6.3.5); other protection choice include:
  - Use of paint or elastomeric/metallic coatings
  - Application of polyurethane (which has shown satisfactory results on thermosets)
  - When high temperature anti-icing measures are used on leading edge parts, upper use temperature of the composite must be considered
- Lightning strike protection methods usually provide adequate erosion protection except on aircraft leading edge faces
- Forward facing edges of laminates which are exposed to the airstream are especially vulnerable and should be protected with edge wrap film or an anti-peel ply (see Fig. 6.2.3)

Marine Environment

Marine environments occur at sea coasts and island areas and, in general, composites are able to withstand this extreme environment beyond the capability of metals. But this environment does attack areas of secondary adhesive bondlines (cold bonding) and mechanically metal fastened joints to weaken composite strength. The marine environment is composed of:

- High humidity
- Highly corrosive salty moisture

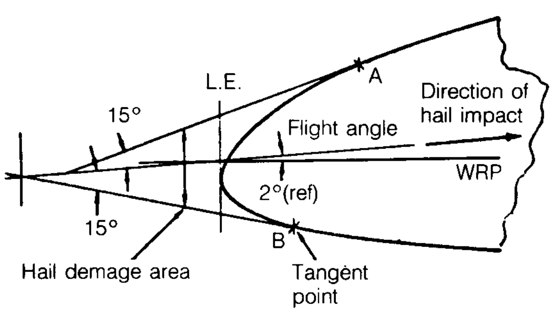


Fig. 6.2.2 Possible Hail Damage Area on The Leading Side Between A and B

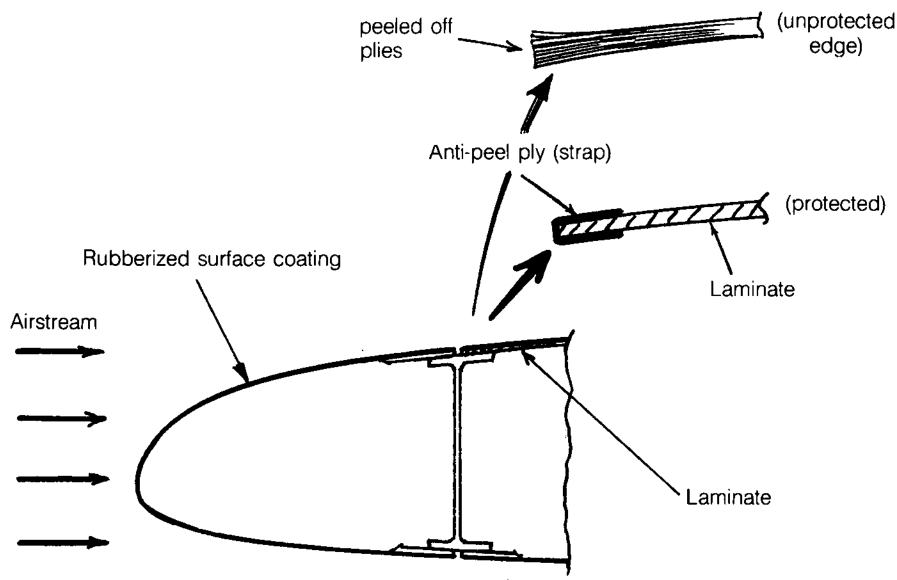


Fig. 6.2.3 Protection of Leading Edge Surfaces

6.3 LIGHTNING STRIKES

There is increased use of composite materials to replace aluminum in primary airframe structures. But composite materials are either not conductive at all or are significantly less conductive than aluminum. Unless protected, composites suffer more damage due to lightning strikes (see Fig. 6.3.1) than counterpart aluminum structures. Also, composite materials allow significant portions of lightning current to flow into onboard systems (e.g., electrical wiring, hydraulic lines, fuel and vent tubes, etc.) and provide less shielding of onboard electronic systems from lightning electromagnetic fields than to metal structures.

Airframe structures require lightning strike protection on exterior or mold line surfaces of the aircraft as defined in the specification listed in Fig. 6.3.2. Aircraft protruding tips, leading edges and trailing edges are the exterior mold line surfaces most likely to be primary lightning strike zones; other airfoil surfaces are secondary strike zones. Both must be conductive to facilitate lightning streamering and the dissipation of static electricity to ground or to static dischargers.

An idealized concept of aircraft lightning protection would be to have the entire exterior surface highly conductive and electrically continuous. This is impossible because of windshield transparency, radome, antenna, and other requirements. In addition, there are weight penalties associated with the idealized protection system. One compromise to idealized lightning protection is to have the exterior surface conductive to a degree that is consistent with system requirements and safe operation. The fuel system require special design attention.

Ref. 6.19 provides information and guidance concerning an acceptable means, but not the only means, of compliance with parts 23, 25, 27, and 29 of the U.S. Federal Aviation Regulations (FAR). Ref. 6.19 describes methods as applicable of preventing the hazardous effects due to lightning, from affecting electrical/electronic systems which perform critical/essential functions. In lieu of following the suggested methods, the applicant may elect to establish an alternative method of compliance that is acceptable to the FAA.

Terms commonly used to describe lightning strikes are given below:

- Attachment Point — A point of contact between the lightning flash and the aircraft.
- Lightning Flash — The total lightning event in which charge is transferred from one charge center to another. It may occur within a cloud, between clouds, or between a cloud and ground. It can consist of or more strikes, plus intermediate or continuing currents.
- Lightning Leader — The leader is the preliminary breakdown that forms an ionized path for charge to be channeled towards the opposite charge center. The "stepped" leader advances in a series of short, luminous steps prior to the first return stroke. The "dart" leader reionizes the return stroke path in one luminous step prior to each subsequent return stroke in the lightning flash.
- Lightning Strike — Any attachment of the lightning flash to the aircraft.
- Lightning Stroke (Return Stroke) — A lightning current surge that occurs when the lightning leader makes contact with the ground or another charge.
- Swept Stroke — A series of successive attachments due to sweeping of the flash across the surface of the airplane by the motion of the airplane.

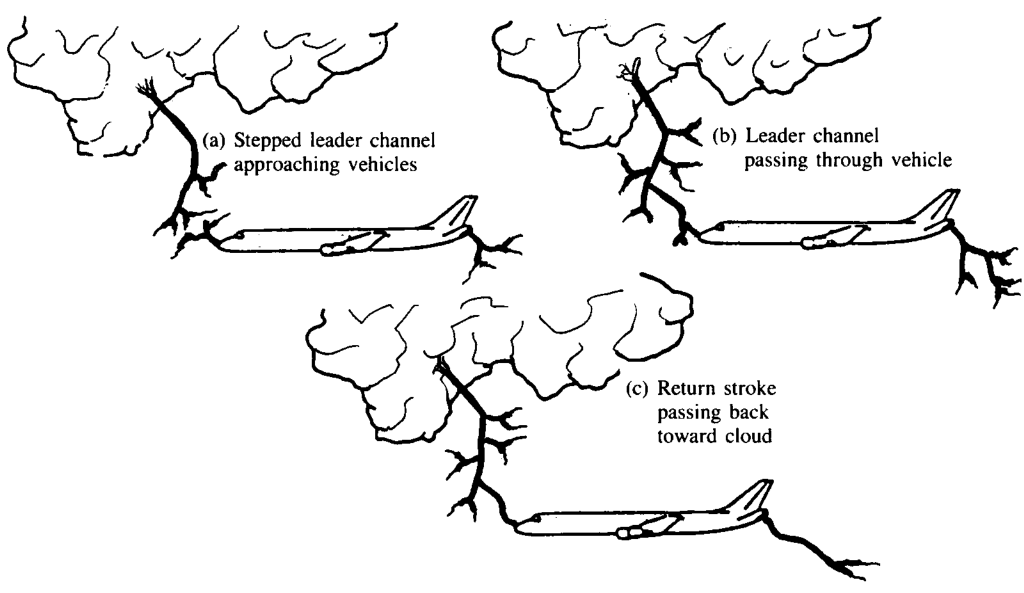
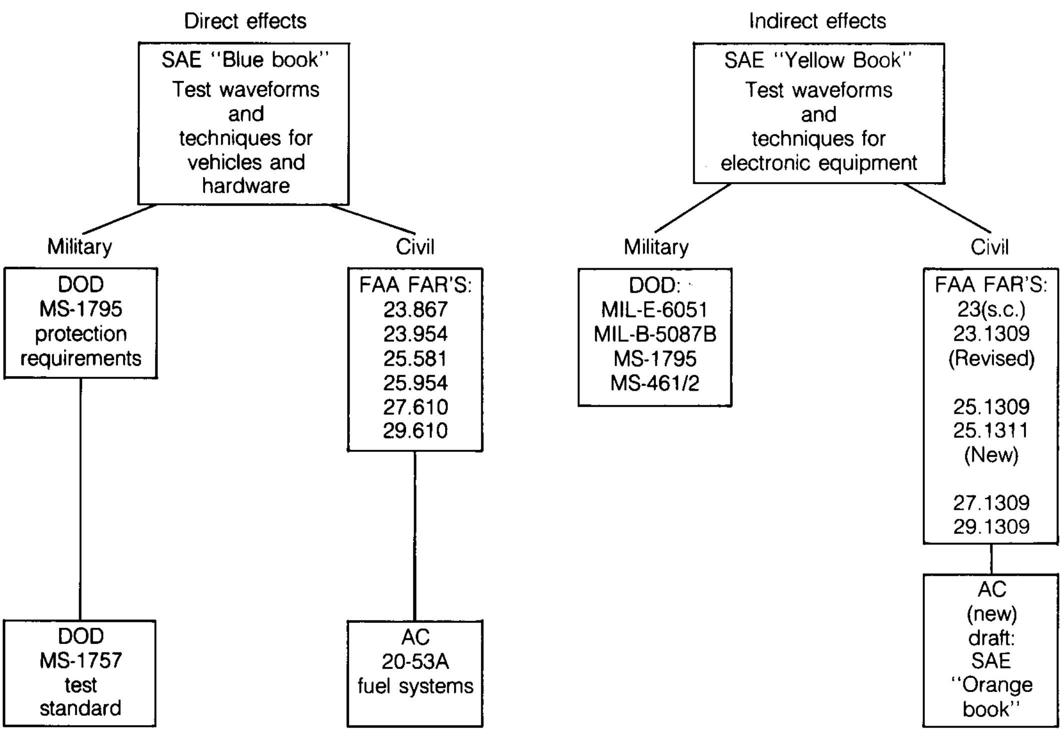


Fig. 6.3.1 Lighting Flash Striking on Aircraft

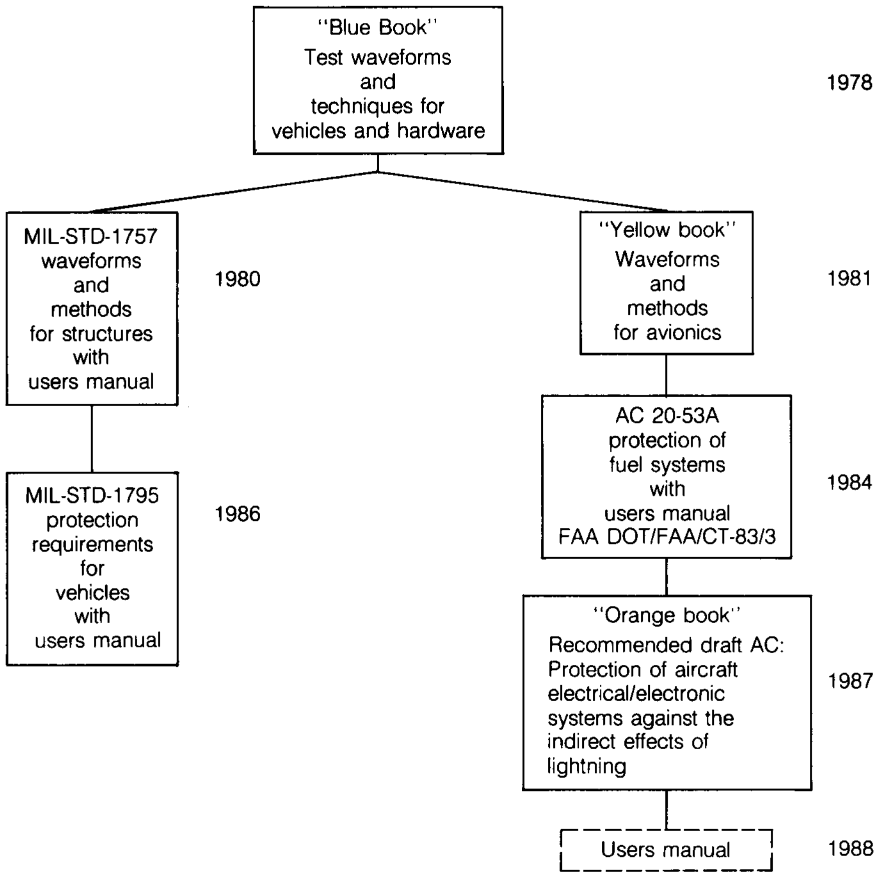
The aircraft can be divided into three lightning zones (Ref. 6.19) as shown in Fig. 6.3.3. These zones are defined as follows:

- Zone 1 — Surface of the vehicle for which there is a high probability of direct lightning flash attachment or exit
  - Zone 1A — Initial attachment point with low probability of flash hang-on, such as a nose
  - Zone 1B — Initial attachment point with high probability of flash hang-on, such as a tail cone
- Zone 2 — Surface of the vehicle across which there is a high probability of a lightning flash being swept by the airflow from a Zone 1 point of direct flash attachments
  - Zone 2A — A swept-stroke zone with low probability of flash hang-on, such as a wing mid-span
  - Zone 2B — A swept-stroke zone with high probability of flash hang-on, such as a wing trailing edges
- Zone 3 — Zone 3 includes all of the vehicle areas other than those covered by Zone 1 and Zone 2 regions. In Zone 3 there is a low probability of any direct attachment of the lightning flash arc, but Zone 3 areas may carry substantial amounts of electrical current by direct conduction between some pairs of direct or swept-stroke attachment points in other zones.



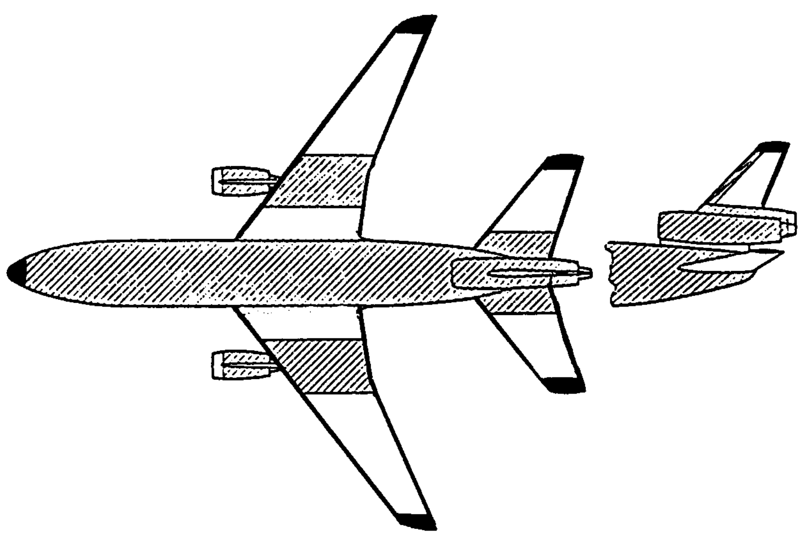
(a) Requirements and specifications

Fig. 6.3.2 Lightning Protection Requirements and Guidelines



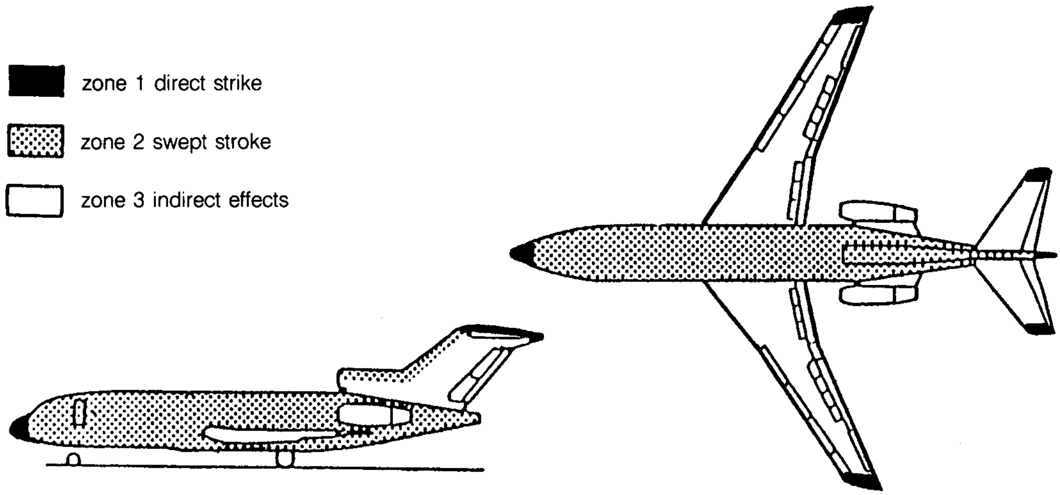
(b) SAE AE4L committee reports (Ref. 6.19)

Fig. 6.3.2 Lightning Protection Requirements and Guidelines (cont'd)



(a) Aircraft with wing mounted engines

Fig. 6.3.3 Lightning Strike Zones — Transport



(b) Aircraft with fuselage mounted engines

Fig. 6.3.3 Lightning Strike Zones — Transports (cont'd)

Aircraft extremities, such as nose, tail cone, wing tips, empennage tips, and engine nacelles are susceptible to lightning attachment. Surfaces aft of the lightning strike attachment points may be contacted by swept strikes as shown in Fig. 6.3.4 and Fig. 6.3.5.

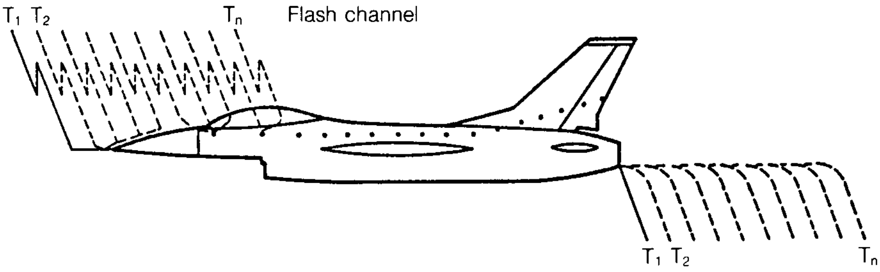
Lightning effects can be divided into direct effects and indirect effects:

- Direct Effects — Any physical damage to the aircraft and/or electrical/electronic systems due to the direct attachment of the lightning channel. This includes tearing, bending, burning, vaporization or blasting or aircraft surfaces/structures and damage to electrical/electronic systems.
- Indirect Effects — Voltage and/or current transients induced by lightning in aircraft electrical wiring which can produce upset and/or damage to components within electrical/electronic systems.

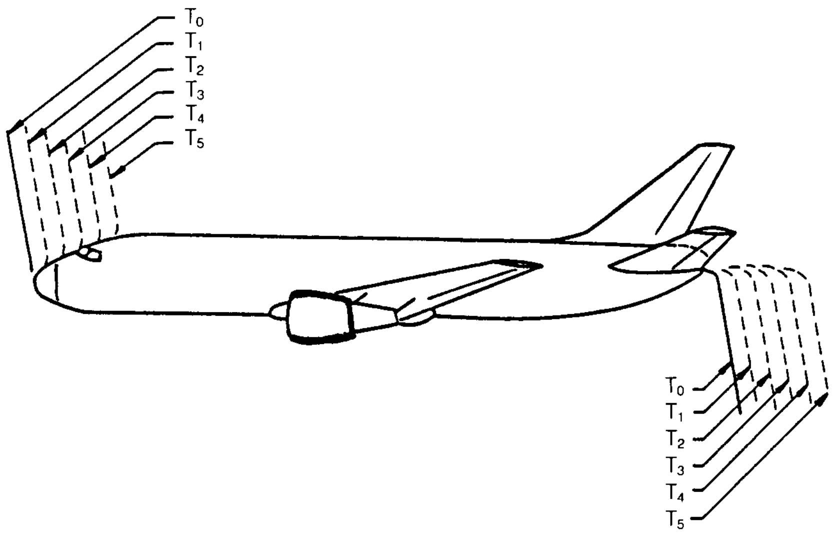
Different components often require different lightning protection methods depending on location of strike (the type of lightning strike zone), material, structural configuration, and interfaces. However, the extent and type of damage frequently can only be determined by testing.

The lightning protection system selected for use in composite applications must satisfy the following requirements:

(1) The system should withstand the mechanical forces involved in dissipating high electrical energy (lightning) loads and provide sufficient conductive surface to substructure continuity for safety-of-flight protection from the electrical wave forms.
(2) Neither the protective system nor its application process should detract from composite material properties.
(3) The system should permit the dissipation and flow of static electricity to a substructure ground or toward static discharges (pigtails, shown in Fig. 6.3.6) and should provide adequate shielding from electromagnetic interference (EMI).



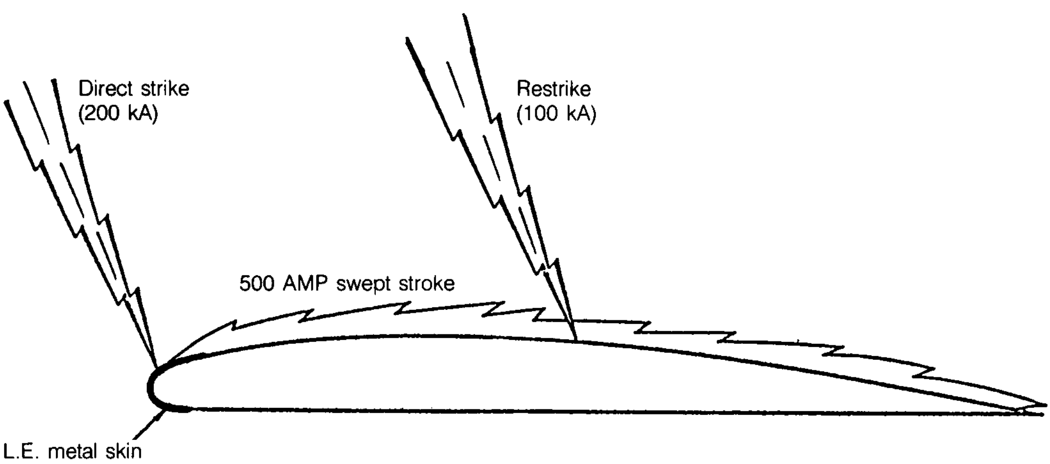
(a) Fighter aircraft



(b) Transport aircraft

By courtesy of Lightning Technologies, Inc.

Fig. 6.3.4. Typical Path of Swept-stroke with Attachment Points



(Metal may required for lighting strike, hail damage and erosion purposes)

Fig. 6.3.5 Lightning Strike on Wing Surface

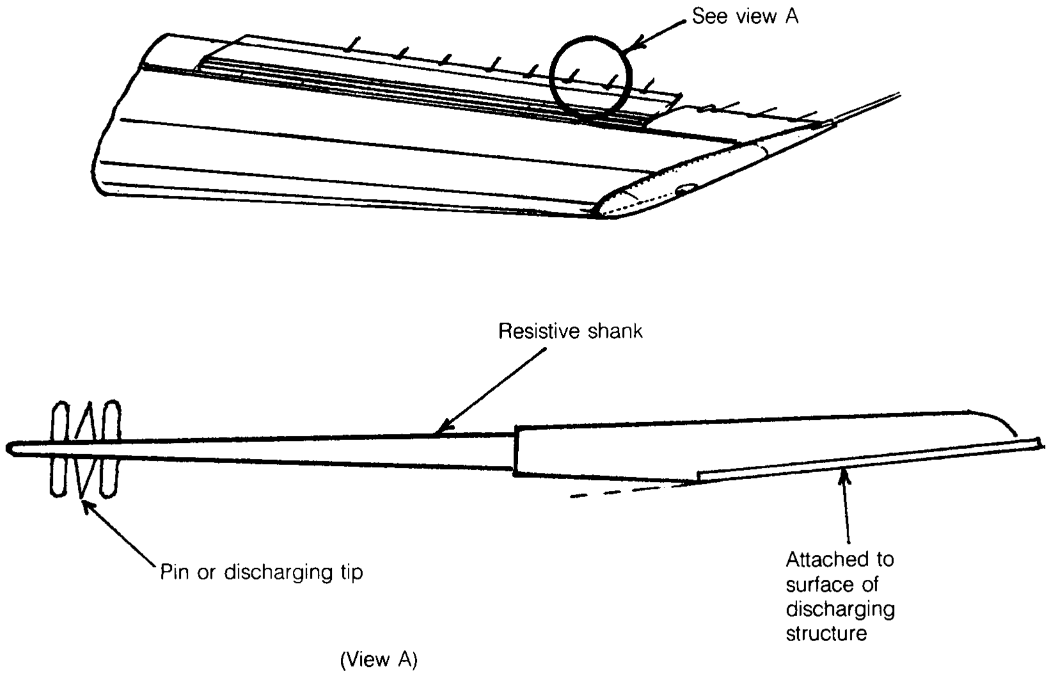


Fig. 6.3.6 Pigtail Precipation Static Discharge on Trailing Edge of Control Surface

(4) The system's conductivity characteristics and the electrical grounding joint should not significantly degrade with time or operational environmental exposure.
(5) The protective surface material system should be repairable, considering flight and ground service exposure conditions, and require a minimum of maintenance.

Areas which may require protection are:

(1) Non-conductive composites (e.g., fiberglass, Quartz, Kevlar, etc.):
   - Do not conduct electricity
   - Puncture danger when not protected
(2) Advanced composite skins and structures:
   - Generally non-conductive except for carbon reinforced composites
   - Carbon fiber laminates have some electrical conductivity, but still have puncture danger for skin thickness less than 0.15 inch (3.81 mm)
(3) Adhesive bonded joints:
   - Usually do not conduct electricity
   - Arcing of lightning in or around adhesive and resultant pressure may cause disbonding
(4) Anti-corrosion finishes:
   - Most of them are non-conductive
   - Alodine finishes, while less durable, do conduct electricity
(5) Fastened joints:
   - External fastener heads attract lightning
   - Usually the main path of lightning trasmission between components
   - Even use of primers and wet sealants will not prevent transfer of electric current from hardware to structure

(6) Painted skins:

- The slight insulating effect of paint confines the lightning strike to a localized area so that resulting damage is intensified
- Lightning strikes unpainted composite surfaces in a scattered fashion causing little damage to thicker laminates

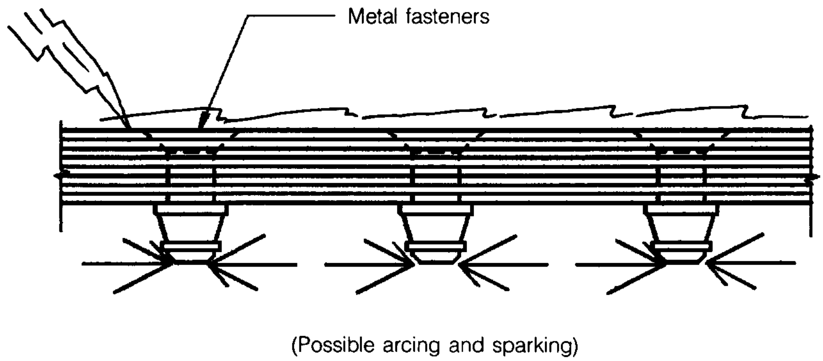
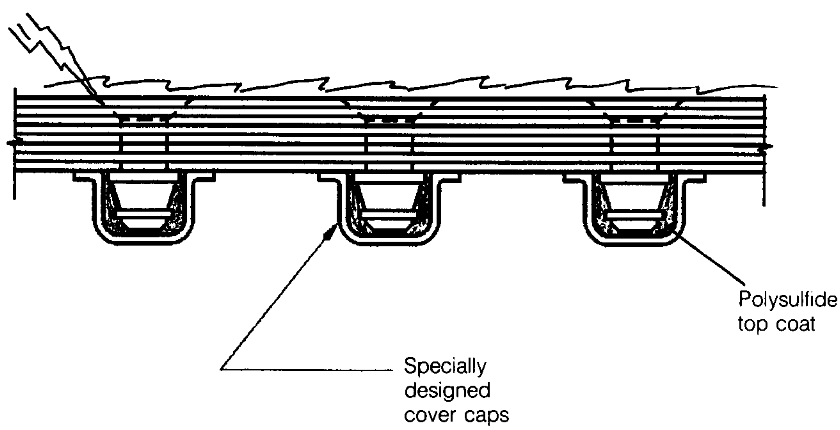
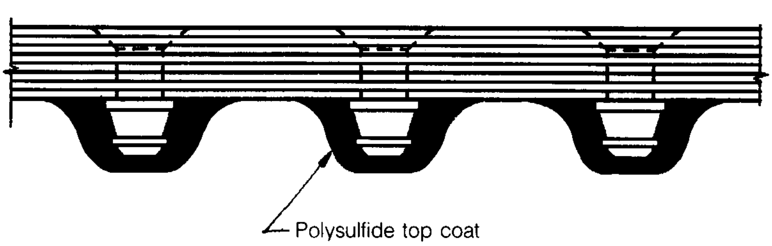


Fig. 6.3.7 Arcing and Sparking Between Metal Fasteners

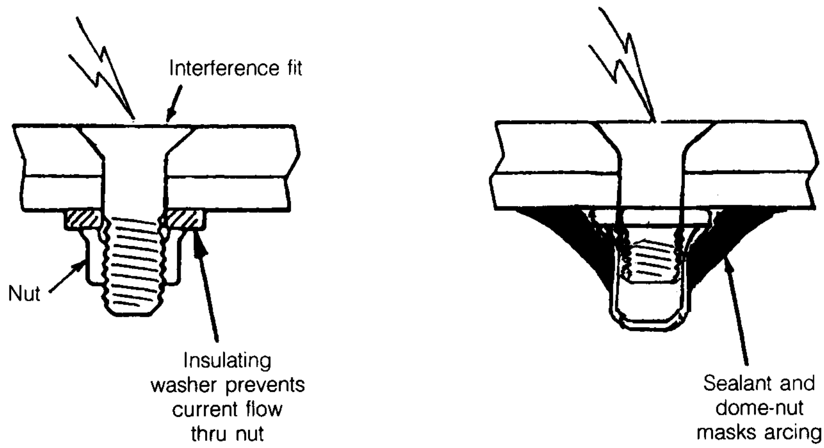


(a) Use plastic cover caps to prevent arcing



(b) Use top coat sealant

Fig. 6.3.8 Methods of Preventing Arcing and Sparking of Metal Fasteners in Carbon Composites



(c) Interference fit fastener with insulating washer

(d) Uuse sealant and Done-nut

Fig. 6.3.8 Methods of Preventing Arcing and Sparking of Metal Fasteners in Carbon Composites (cont'd)

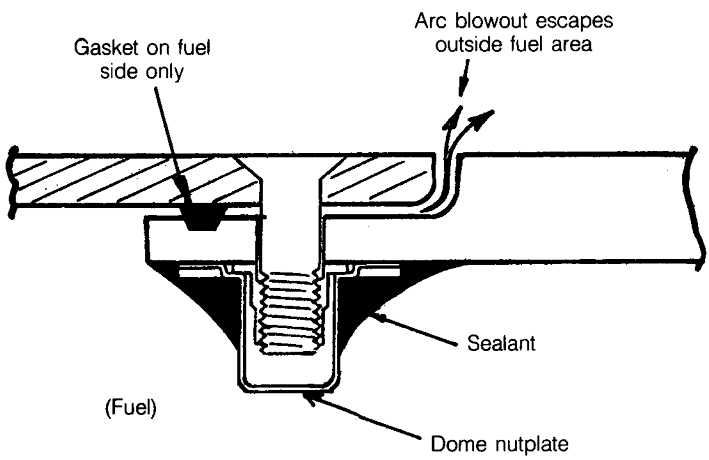


Fig. 6.3.9 Lightning Protection of Fuel Tank Access Door Seal

(7) Integral fuel tank:

- Dangers are melt-through of metal fasteners or arc plasma blow by between fasteners and the resulting combustion (see Fig. 6.3.7); some preventative measures are shown in Fig. 6.3.8
- Lightning protection for access doors is shown in Fig. 6.3.9
- Wing fuel tank skins will provide adequate protection against ignition of fuel vapors, if skin thicknesses are electrically equivalent to an aluminum thickness of 0.08 inch (2.032mm) (swept stroke studies indicate that 0.08 inch aluminum thickness would be adequate for conventional airframes)
- Consider use of bladders in aircraft wings which carry fuel to eliminate possibility of metal fastener arcing
- Use composite material fasteners

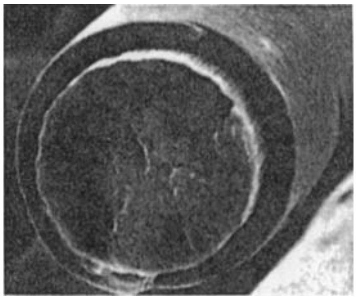
- Lightning tests are generally required to demonstrate the satisfactory suppression of internal sparking or arcing in the fuel tank area (Zone 2)
- Fuel vents — Fuel vapor in vents represents a potential fuel combustion explosion hazard

Lightning Protective Methods for Conductive Carbon Composite Materials

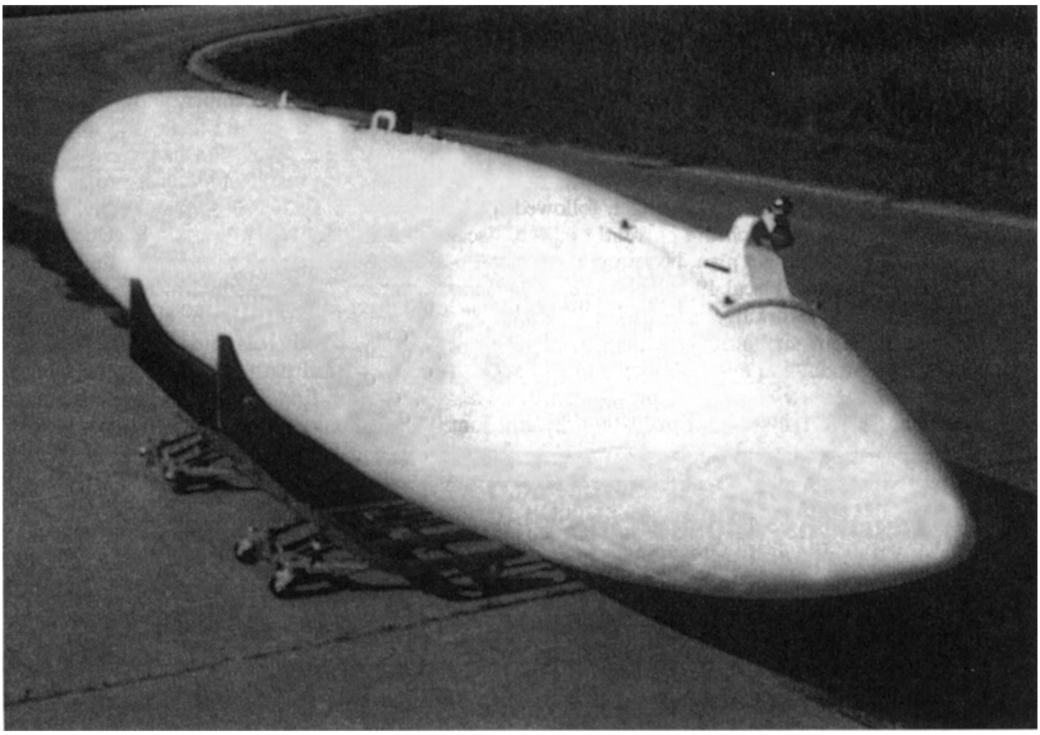
(1) Woven wire mesh:
- Aluminum wire diameter up to 0.008 inch (0.203 mm) located on 0.125 inch (3.175 mm) centers (0.0128 lb/ft^2)
  - Aluminum incompatible with carbon composites
  - Fiberglass scrim cloth may be sandwiched between mesh and carbon to prevent corrosion but this degrades protection mechanism by impeding current flow into carbon
- Protects carbon composites from particle erosion
- Weight penalty
- Difficult to install on compound contoured surfaces
- Mesh cocured with laminate
- Increases erosion resistance
- Must be used on outermost surface layer only (do not use multiple layers because this will result in interlaminate explosion)
- Can be painted

(2) Aluminum flame or arc spray [0.006 inch (0.152 mm) to 0.008 inch (0.203 mm) thick]:
- Not desirable for carbon composite because aluminum and carbon are galvanically incompatible
- Fiberglass scrim cloth may be sandwiched between mesh and carbon to prevent corrosion but this degrades protection mechanism by impeding current flow into carbon
- Coating weight and quality is operator-dependent
- Finish is irregular if sprayed on surface of material
- Crazing may occur when applied to large areas
- Can be painted

(3) Nickel coated fibers:
- Nickel is electrodeposited on graphite filament as shown in Fig. 6.3.10
- Lower conductivity than aluminum
- Minimal weight increase
- Substituted for one layer of material in layup
- Can only be applied at surface layer
- May be painted
- Small loss of strength
- Expensive



(a) 0.5 Micron thick nickel layer on about a 7 micron graphite fiber



(b) Filament-wound composite fuel tank is protected against lighting strikes by nickel-coated graphite fibers

By courtesy of American Cyanamid Co.

Fig. 6.3.10 Nickel Coated Fibers

(4) Aluminum interwoven wires [0.004 inch (0.102 mm) wires and 200 wires/inch]:
- Wires must be imbedded in matrix and kept dry, otherwise possible galvanic corrosion between wires and carbon composites
- Does not protect against indirect effects
- Outermost ply only
- Works well under painted surfaces

(5) Conductive paints:
- High metal (copper or silver) content necessary for adequate conductivity
- Cannot handle the higher levels of electric currrent
- Works on the principle of diverting flash current over material surface
- Cannot be used in proximity to other conductive elements
- Can be painted
- Least effective method
- Should not be used for conductive composites (e.g., carbon, boron, etc.) because it causes greater damage

(6) Aluminum bar diverter with use fasteners:
- Bar diverters may be used to transfer lightning current across Zone 3 regions and they must be properly located to avoid sharp bends of corners, which may be damaged by the magnetic forces during the transfer of lightning current. Fig. 6.3.11 shows a example of bar diverter application
- Able to withstand first return stroke
- Reduced efficiency when painted

(7) Conducting polymers (not commercialized) — Conducting polymers have been around since the mid-1970s and but are still in developmental stage. They have the potential to protect against lightning strikes and dissipate energy over the surface of the material, as shown in Fig. 6.3.12.

Lightning Protective Methods for Non-conductive Composite Materials

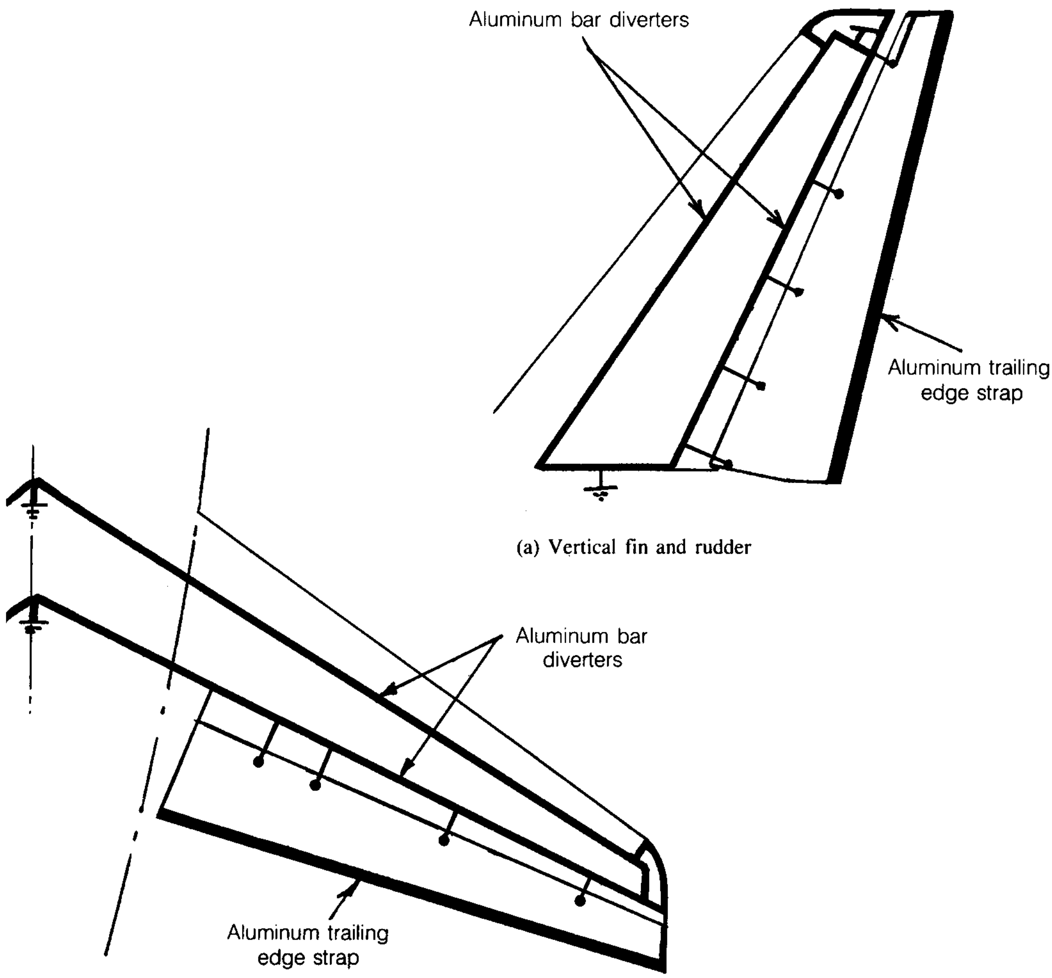
Non-conductive materials such as fiberglass, Kevlar, Quartz, etc. must be protected from electromagnetic waves (an indirect effect) produced by lightning strikes:

(1) For "transparent" applications (e.g., radome radar fairings, etc.) consider the following:

(a) Aluminum bar diverter [diverter strips; 0.5 inch (12.7 mm) × 0.25 inch (6.35 mm)] and fasteners (#6-aluminum or #8-copper) with spacing 6 to 8 inch:
- The nose radome (see Fig. 6.3.13) is not conductive and bar diverters used on the exterior surface of the radome tend to shield the interior components.
- Reduced efficiency when painted
- May hinder dispersion of electromagnetic waves, although weather radar is not affected
- Commonly used on transport radomes

(b) Segmented diverter (shown in Fig. 6.1.14)
- Easily installed
- Transparent to electromagnetic waves
- Not as effective as aluminum bar diverter
- Cannot be painted
- Commonly used on radomes of fighter aircraft and some commercial aircraft use also

(c) Dielectric film such as Lexan, Kapton, Ultem, etc.
- Particle impact reduces insulating ability
- Used on interior applications where not affected by particle impact



(a) Vertical fin and rudder

(b) Horizontal tail and elevator

Fig. 6.3.11 Lightning Strike Protection on A320 Tail Planes-Use Aluminum Bar Diverter Bolted Around The Composite Structures

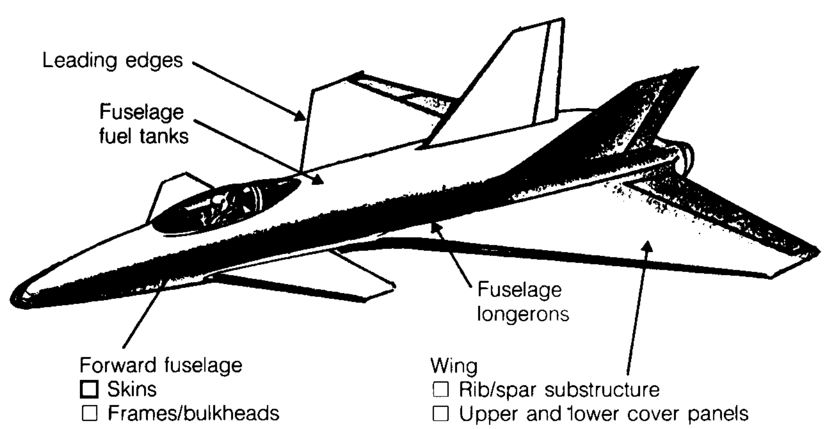
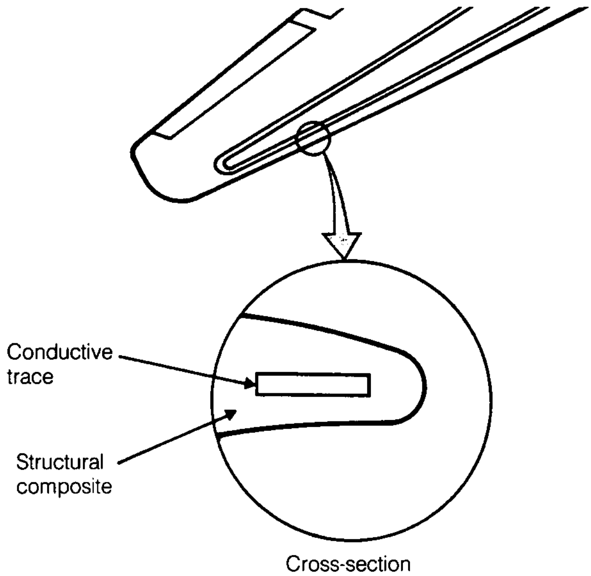
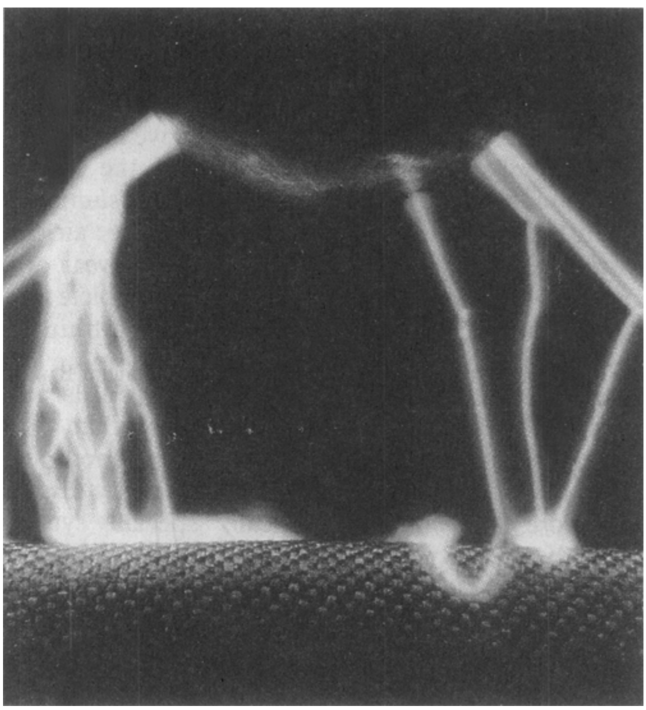


Fig. 6.3.12 Conductive Polymers



(a) Artist's concept of future advaced fighter features using conductive polymers



(b) Aluminum lightning dissipates over the surface of conductive polymers

By courtesy of Lockheed Aeronauticaly Systems Co.

Fig. 6.3.12 Conductive Polymers (cont'd)

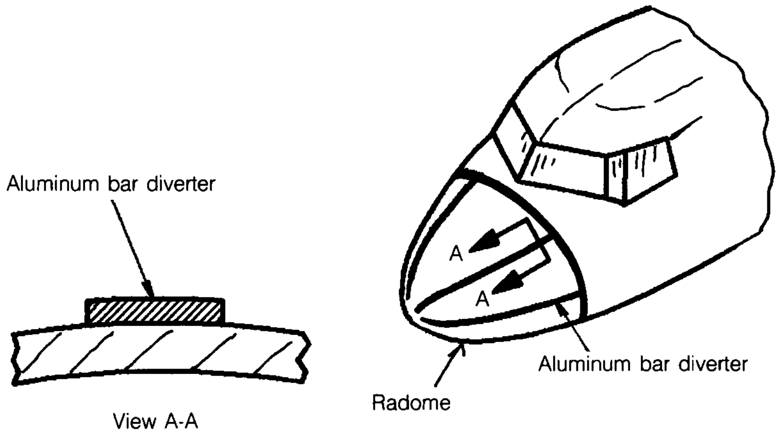


Fig. 6.3.13 Aluminum Bar Diverter Applied on Radome

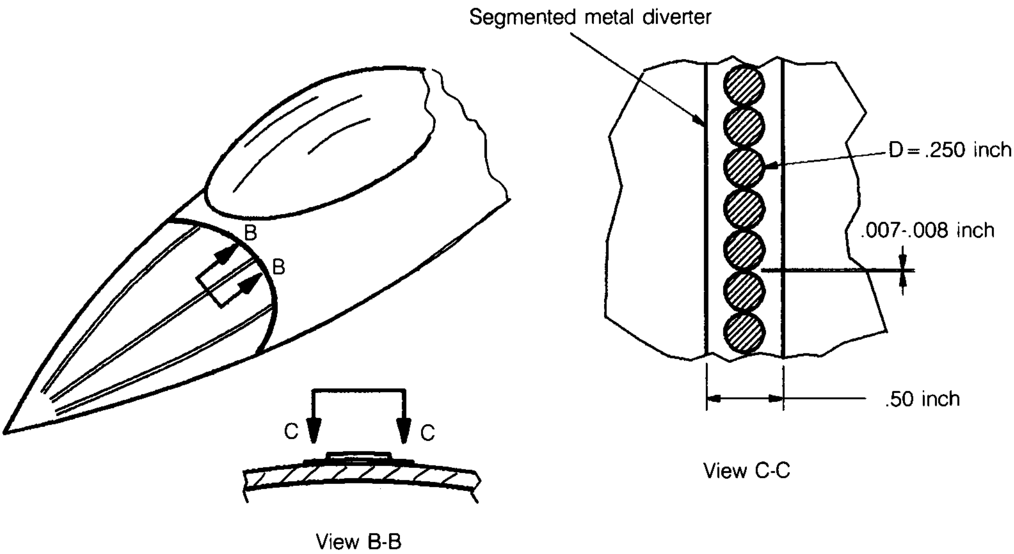


Fig. 6.3.14 Segmented Metal Diverter

(2) For "non-transparent" applications consider these protective measures:
  (a) Aluminum foil [0.005 inch (0.127 mm) thick]:
    - Environmental seal of composite surface
    - Uniform surface conductivity
    - Surface material completely replaceable
    - Foil stock width limiations
    - Difficult to install on compound contours
    - Poor repairability
    - Poor handleability

(b) Aluminum flame or arc sprayed [0.005 inch (0.127 mm) thick]:
- Very effective protection
- Coating weight (approximately 0.1 lb/ft^2) and quality is operator-dependent
- Other characteristics similar to those listed under item (2) of "Lightning Protective Methods for Conductive Carbon Composite Materials"
- Can be applied to compound contours

(c) Aluminum woven wire mesh [0.004 inch (0.102 mm) and 200 wires/inch]:
- Weight penalty is approximately 0.045 lb/ft^2
- Other characteristics similar to those listed under item (1) of "Lightning Protective Methods for Conductive Carbon Composite Materials"

(d) Expanded copper or aluminum mesh (see Fig. 6.3.15):
- Consists of solid foil that is slit, expanded and rolled flat
- Due to long continuity, precludes sparking and gives much better conductivity than woven mesh
- Drapeable and works well on compound contours
- Works best if applied by prepregging to first ply and cocured with laminate
- Mesh stock width limitations
- Use of paint concentrates damage
- Poor handleability
- Inhibits radar and radio waves

(e) Metallized cloth (aluminum or nickel plated fibers or cloth):
- Aluminized fiberglass (see Fig. 6.3.16) or metallized Kevlar
- Other characteristics similar to those listed under item (3) of "Lightning Protective Methods for Conductive Carbon Composite Materials"

(f) Conductive paint (dope with carbon, silver, or copper particles):
- Characteristics similar to those listed under item (5) of "Lightning Protective Methods for Conductive Carbon Composite Materials"

Electromagnetic Interference (EMI)

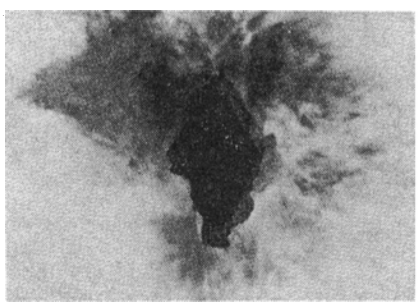
The shielding of sensitive and critical electronic equipment from external electromagnetic interference (EMI) is of vital importance in aerospace systems for safe flight operation (see Fig. 6.3.17). This is most directly accomplished by surrounding such equipment with an electrically conductive shell (metallic structure performs this function automatically).

- Organic composites (excluding metal matrix composites) are not adequate conductors or adequately grounded or interconnected to absorb by induction the incoming electromagnetic radiation
- Minimal conductive coatings, such as conductive paints, may not provide adequate EMI shielding
- Protective methods used to safeguard against lightning strikes usually serve as effective EMI shielding
- Two basic methods exist for protecting electronics from lightning:
  — Simply shield them with metal
  — Electrically isolate them from the induced charge

	Copper	Aluminum
Base material	110 Electrolytic Tough Pitch Annealed Copper	1145 Series Electrical Grade Annealed Aluminum
Conductivity vs. Silver Standard	101%	59%
Resistivity, micro ohm/cm	1.71	2.89
Minimum Elongation of Screen Material	100%	20%
Tensile Strength, ksi		20
Thickness, Inches Minimum Maximum	 0.003 0.015	 0.003 0.015
Minimum Weight lb/ft^2	0.03	0.015
Strand Width, Inches Minimum Maximum	 0.003 0.020	 0.003 0.020
Open Area, %	15-85	15-85
Roll Length, ft.	650	650
Std. Roll Widths, in.	13 & 31	13 & 31

(Source: Astroseal products Manufacturing Corp.)

Fig. 6.3.15 Lightning Strike Screen (non-woven) Materials



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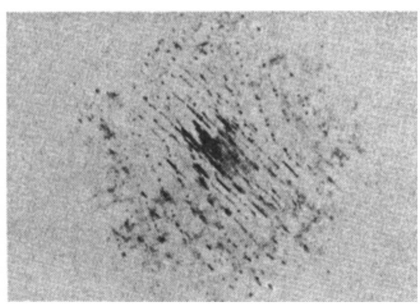


(protected)

Fig. 6.3.16 A Single Ply of Thorstrand Aluminized Glass Cloth Offers Significant Protection From Lightning

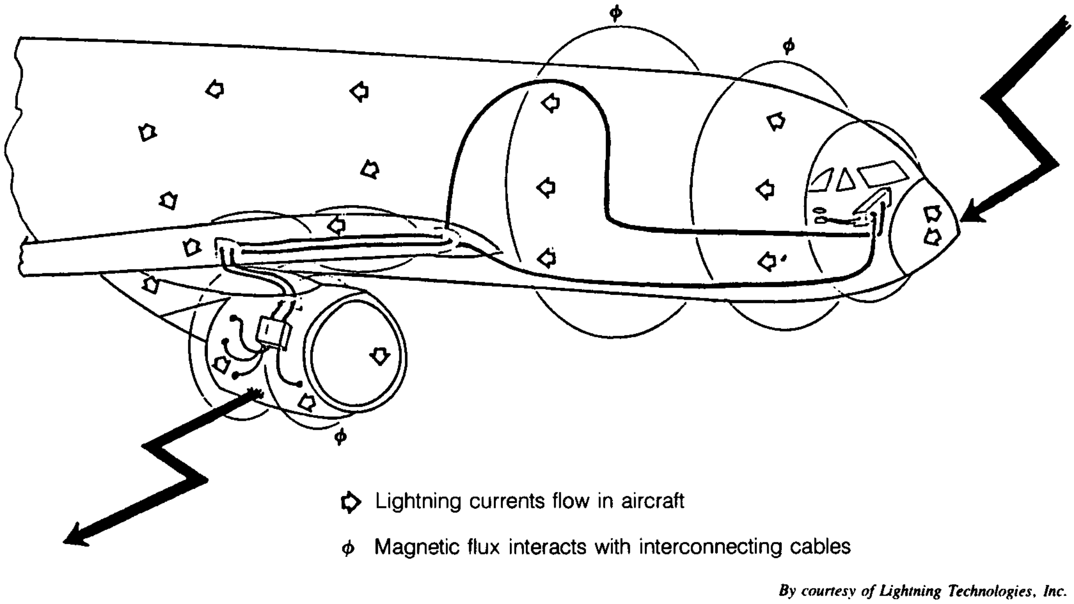
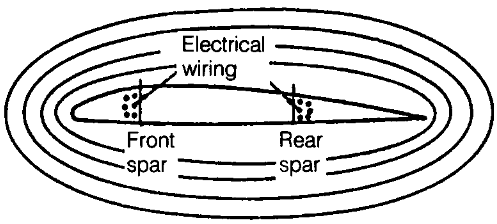
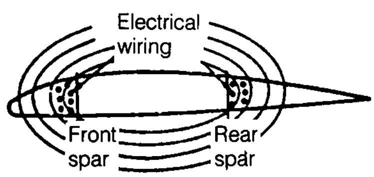


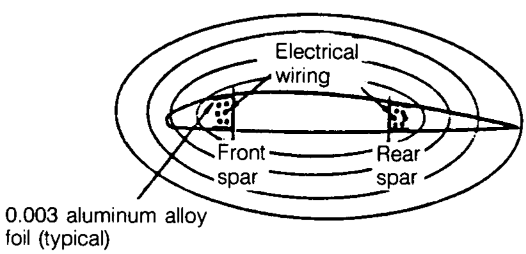
Fig. 6.3.17 Lightning Intervaction with Electronic Systems



(a) Aluminum alloy leading edge & trailing edge



(b) Composite leading edge & trailing edge



(c) Composite with aluminum alloy foil

Fig. 6.3.18 Composite Edge Panels Affect Electromagnetic Field

Composites (usually Fiberglass or Kevlar) are being used extensively for fixed leading and trailing edges (areas of minimum skin thickness). This presents a problem when electrical wiring needs to be installed in these areas. An electromagnetic field around a structure results, as shown in Fig. 6.3.18, from the flow of current from a lightning strike. If the leading and trailing edge are all-metal alloy, no problem exists. Lightning strike inducing higher voltage in the wiring may result in damage to engine instruments. Also, overloading wiring to fuel quantity probes in the integral tank may result in arcing to the structure. These problems can be solved by:

- Running wiring in conduits
- Putting fuel quantity probe wiring inside the integral tank
- Protecting the engine or control wiring by applying 0.003 inch (0.0762 mm) self-bonding aluminum alloy foil onto the composite skin panels to increase the electromagnetic field, as shown in Fig. 6.3.18(c).

6.4 GALVANIC CORROSION

The aircraft industry has vast amounts of data gained from history and research on corrosion to design out corrosion and to manufacture corrosion resistant airframes (see Ref. 6.9). This Section describes the most pronounced galvanic corrosion problems faced today and the means being employed to implement prevention into design and manufacturing practice.

Galvanic corrosion occurs where two materials from different groups in the galvanic series, as shown in Fig. 6.4.1, are in contact in the presence of moisture. This type of corrosion is usually accompanied by a buildup of corrosion products in the contact area. Corrosion progresses more rapidly the farther apart the materials are in the galvanic series. Material located toward the anodic end of the table will corrode sacrificially. For example, aluminum will corrode when in contact with carbon (or graphite) composites. This is especially a problem with the metallic fasteners generally used in assemblies (see Fig. 5.2.7 in Chapter 5).

Galvanic corrosion can be reduced by insulating the materials in the areas of contact and by applying protective coatings on both materials.

Recommendations are given below:

(1) Carbon composites in contact with metals
- Carbon will induce corrosion when in contact with aluminum when moisture exists, unless moisture intrusion is prevented
- Keep carbon composites out of electrical contact with any adjacent metals
- Carbon is highly cathodic and will severely attack aluminum and cadmium unless a protective ply of inert cloth (e.g., fiberglass, Kelvar, etc.) is used between them and/or protective paint or sealant is used

(2) If an aluminum part is to be used in contact with carbon components, all aluminum and carbon parts should be processed as follows prior to assembly:
  (a) Method A:
  - Anodize aluminum parts per MIL-A-8625, Type II
  - Finish external surfaces of both aluminum and carbon parts per MIL-F-18264:

- Two coats epoxy primer per MIL-P-85582
- Two coats white polyurethane enamel per MIL-C-83286

(b) Method B:
- Cure a layer of fiberglass ply to the composite interface
- Seal edges of composites

(3) Boron composites in contact with metals
- Corrosion does not occur when elemental Boron contacts metal
- The Boron tungsten core fiber can be a corrosion cell if an electrical connection exists between the tungsten core and other metals such as a fasteners (anodic materials)

(4) Fasteners used with composites
- Use fasteners of corrosive resistant materials (e.g., titanium, corrosion resistant steel, etc.) for carbon composites (see Fig. 5.2.7 in Chapter 5)
- Do not use cadmium-plated fasteners in carbon composites
- Install fasteners wet with corrosion inhibiting sealant

Most Active
(anodic)

Increasing tendency to corrode

- Magnesium alloys
- Alclad 7000 series aluminum alloys
- 5000 series aluminum alloys
- 7000 series aluminum alloys
- Pure aluminum and alclad 2000 series alloys
- Cadmium
- 2000 series aluminum alloys
- Steel and iron
- Brass and bronze alloys
- Stainless and heat resistant steels
- Titanium
- Nickel and nickel alloys
- Graphite composites

Least active
(Cathodic)

Fig. 6.4.1 Galvanic Series of Aircraft Materials

6.5 OTHER ENVIRONMENTAL CONCERNS

(1) Solvents and chemical resistance:
Composite properties may degraded by long time exposure to aircraft fluids/chemicals:
- Application of flexible polyurethane paint is recommended
- Aircraft fuel is absorbed by the composite matrix and causing weight gain and reduction of strength.
- Hydraulic fluid does not usually affect most composites (Skydrol does)
- Paint strippers will attack thermoset and thermoplastic composites
- MEK cleaning fluid dissolves some thermoplastic composites

(2) Acoustic exposure:
When panel structures are located in the following areas, be aware that premature failure could be caused by exposure to acoustic environment:

- Cavity areas of bomb-bay, landing gear storage, etc.
- Near or close to engine areas
- Use sandwich or honeycomb panels
- Use bonded-joints instead of fasteners to eliminate stress concentration

(3) Nuclear exposure:

Protection from the following effects should be considered when aircraft structures are exposed to a nuclear explosion:

- Dynamic overpressure — a dynamic airload
- Thermal shock — could result in significant material degradation if panel surface has no protection or coating
- Nuclear radiation — in general, this does not cause the degradation of material for currently used composites; exposure to radiation should be considered in the development of future composite materials

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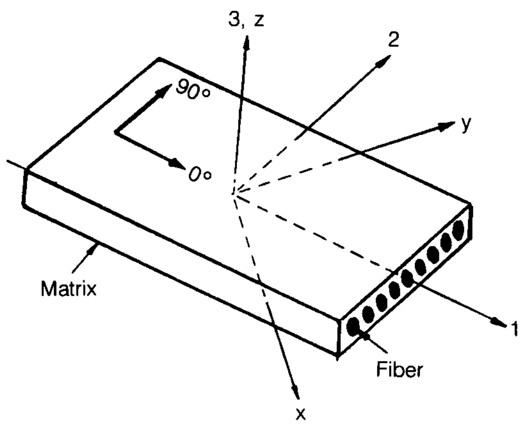
Chapter 7.0

LAMINATE DESIGN PRACTICES

7.1 INTRODUCTION

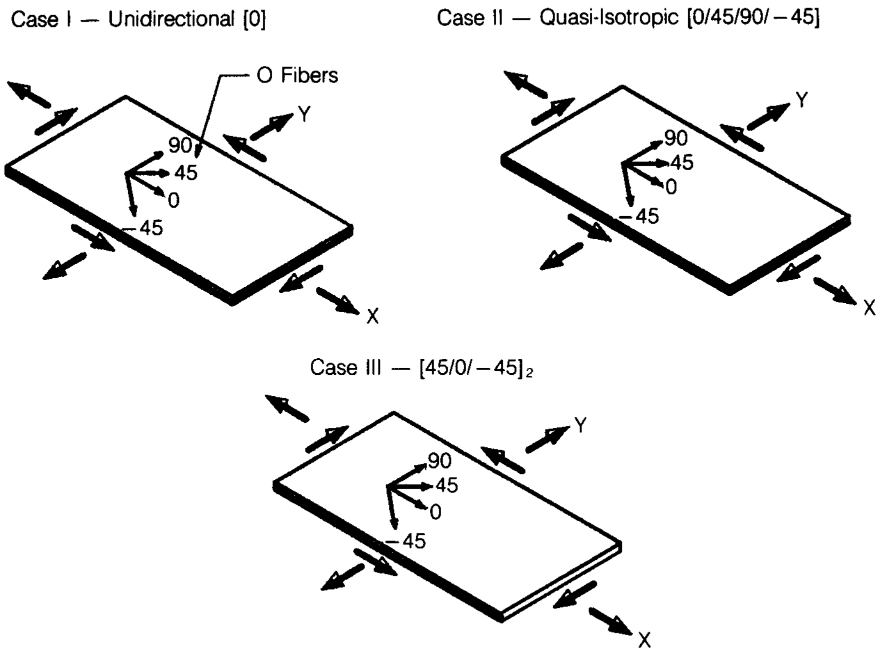
Composite materials are uniaxial in their single-ply site state as shown in Fig, 7.1.1, having very high mechanical properties along their longitudinal axis, and low properties along their transverse axis (tape only). This is the primary difference, from a structural design and analysis standpoint, between composites and metals (see Fig. 7.1.2 and Fig. 7.1.3).

Metals are nearly homogeneous and isotropic in nature, and their reaction to an applied load can be defined by knowing two of the three basic elastic constants, the modulus of elasticity (E), the modulus of rigidity (G) and the Poisson's ratio (ν). On the other hand, a basic unidirectional lamina, or any balanced (implies that for every 45° ply, there exists a −45° ply in the laminate) and symmetric (requires mirror image ply stacking about the midplane) laminate, is orthotropic in nature, having three mutually perpendicular planes of elastic symmetry. For planar applications, these types of materials can be defined by four of five basic elastic constants for orthotropic materials. It should be noted that there are twice as many independent planar elastic constants for orthotropic materials as there are for isotropic materials because of the different properties in the planes of symmetry (see Fig. 1.3.1).



(Note: 1, 2, 3 are Lamina coordinates; x, y, z are laminate coordinates)

Fig. 7.1.1 Lamina Axes Rotation



	F_x (Ksi)	E_X (Msi)	F_Y (Ksi)	E_Y (Msi)	F_{XY} (Ksi)	G_{XY} (Msi)
CASE I	1.0	1.0	1.0	1.0	1.0	1.0
CASE II	.38	.4	6.8	5.26	3	2.67
CASE III	.6	.58	4.5	2.3	4	3.6

Fig. 7.1.2 Laminate Strength Variation Versus Ply Angle Orientation

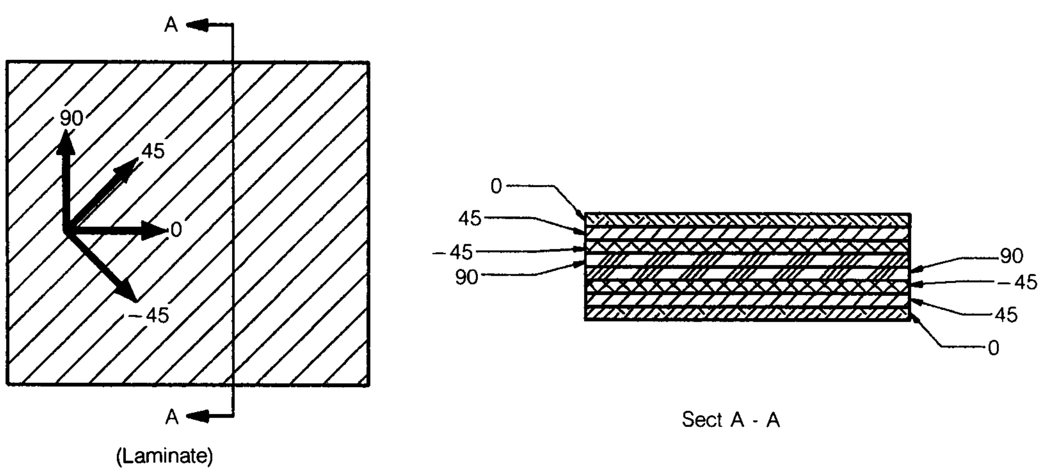


Fig. 7.1.3 Effects of Stacking Sequence

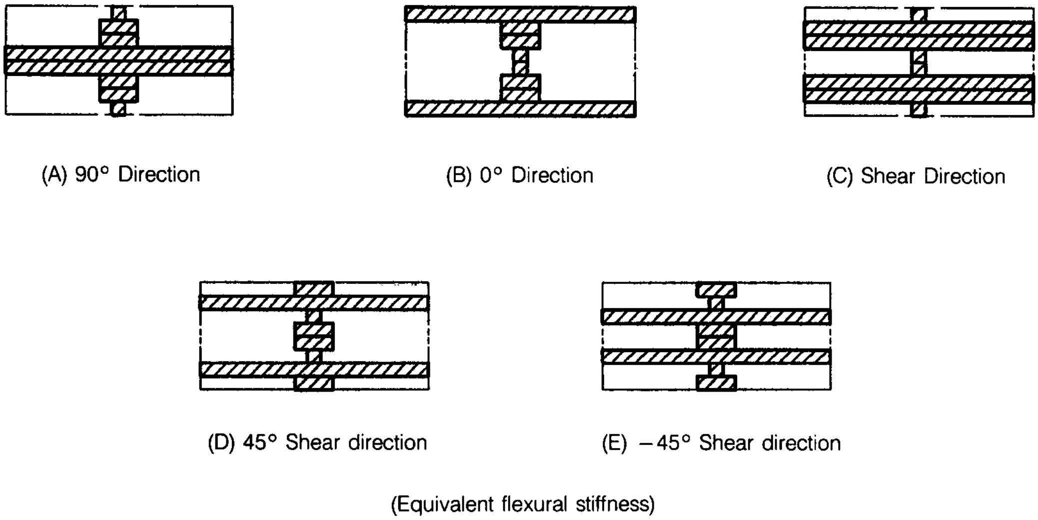


Fig. 7.1.3 Effects of Stacking Sequence (cont'd)

For many composite applications (e.g., the forward swept wing of X-29A, shown in Fig. 1.1.8), the laminate is not even orthotropic, but is anisotropic, as shown in Fig. 7.1.4. This occurs when an orthotropic laminate is loaded in a direction which does not coincide with one of the principal axes, or when the laminate layup is symmetric but not balanced about a principal reference axis. A $[0/\pm45/90]_s$ laminate is balanced and quasi isotropic, while a $[0/\pm45]_s$ laminate is orthotropic. This type of laminate requires six elastic coefficients for definition. As a general rule, all laminates should be symmetrically laid up about their midplane; coupled laminates should be avoided. Fig. 7.1.5 shows the difference between the different types of laminates, including the differences a symmetric laminate and one which is not symmetric.

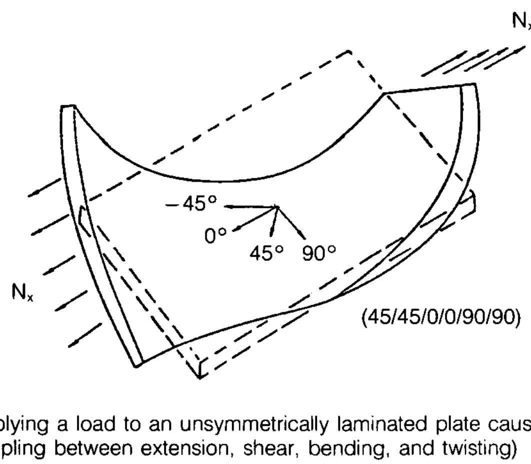


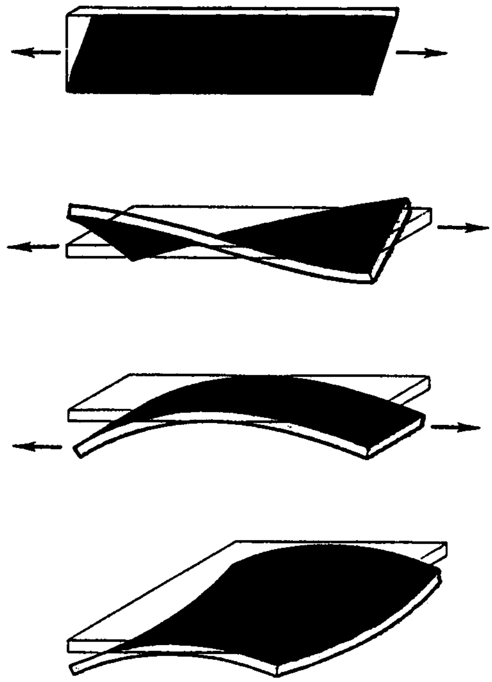
Fig. 7.1.4 Anisotropic Laminate Behavior

All plies at $\theta°$. Axial load results in stretching-shearing behavior.

Two plies at $\pm\theta$ (any angle). Opposing shear deformations in the plus and minus plies result in stretching-torsion interaction.

A 0°/90° stacking. This arrangement bends under pure tension because the modulus-weighted centroid is not coincident with the geometric centroid, resulting in an offset load path.

Another 0°/90° stacking. Because of different thermal expansion characteristics in each layer, this stacking deforms into a "saddle" when heated.



(sketches show how simple loads result in unusual deformations because of coupling action. With balance and symmetry present, these effects disappear)

Fig. 7.1.5 Effective of Stacking Sequence on Deflection of Laminates

The directional nature of composite laminae provides the ability to construct a material which can meet specific loads and/or stiffness requirements without wasting material by providing strength and stiffness where they are not needed. If the design requirement is simply to provide axial strength or stiffness, a high percentage of the material should be unidirectionally oriented. If this material is adhered to restraining members, theoretically all of the fibers may be so oriented [see Fig. 7.1.6(a)]. If the composite material is unconfined, it is wise to provide a nominal amount of transverse reinforcement to account for any off-axis (refers to rotation about any one of the laminate axes) loading that may occur, whether during fabrication or by induced loading and to reduce the Poisson's ratio effects.

If shear loading or shear stiffness is the primary design consideration [see Fig. 7.1.6(b)], then most of the material should be oriented at $\pm45°$ to the longitudinal axis, as this provides the highest shear properties. However, care must be taken to evaluate any loading in the longitudinal or transverse directions, since the strength in these directions is quite low. Conditions which would not normally be critical in metal design may approach or exceed the strengths available in these directions when using a pure [$\pm45°$] layup. This consideration may necessitate the inclusion of at least a minimal number of 0° and/or 90° laminae.

An example of an unrestrained application would be any plate or skin, whether or not it is an elastic base, or a stiffener application. A rule-of-thumb is to make the number of laminae in the transverse direction equal to at least 10% of the total number of laminae in the laminate part.

In addition to the pure axial or pure shear case just mentioned, there are many applications which require the ability to withstand a combination of loadings [see Fig. 7.1.6(c)]. Although it is possible to determine an optimum orientation sequence for any given loading condition, it is more practical from a fabrication standpoint to limit the number of orientations (see Fig. 7.1.7) to a few specific families (e.g., 0°, ±45° and 90°) which can then be characterized by tests.

The anisotropic nature of composite materials, while allowing the engineer to tailor material more closely to the design requirements, imposes the problem of selecting the proper orientation for the application. This is a consideration which does not arise in metal design, and the engineer must be aware that the traditional methods of design and analysis have to be developed to higher orders of refinement for anisotropic materials, not only to provide a basis for selecting the proper orientation, but even for defining stresses and margins of safety (see Fig. 7.1.8).

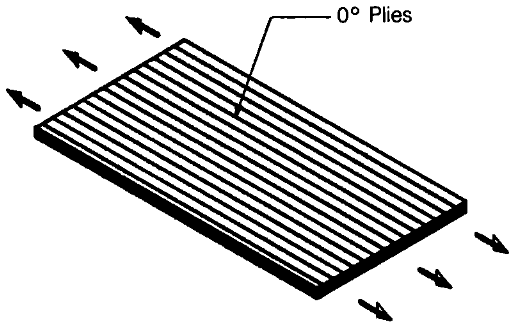
One of the major differences in composite analysis, as opposed to metal, is that strain is the major concern and not stress. Composite structures are made up of different plies, and each ply will be stressed at a different level because the ply's elastic modulus is dependent upon the ply orientation. It is engineer's responsibility to determine the best ply orientation for the various loading conditions (see Fig. 7.1.9) following basic laminate requirements.

In the design of large structures, one of the basic ground rules is to establish the ultimate gross area cut-off strain (or stress), as shown in Fig. 7.1.10(a), when designing in tension and compression-critical areas. This cut-off automatically covers many design considerations, such as high local strain areas, joints of various kinds, and structural integrity in terms of impact damage and fracture. In the application of composite material to structures, the allowable levels (expressed in strain instead of stress) are low because of the following limits:

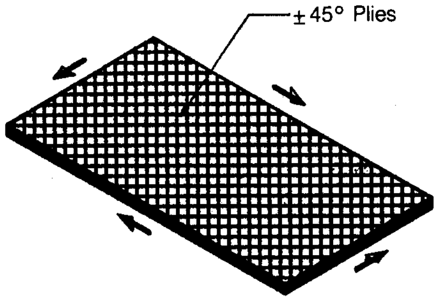
- Tolerance for impact damage (this is the dominant failure mode in compression for some materials)
- Flaw growth resistance
- Stress concentration associated with cutouts, fastened joints, etc.
- Reduced strength in hot/wet conditions

Currently, these factors restrict design ultimate gross area strains to about 50% of the unnotched, undamaged composite material failure strain, depending on loading and laminate orientation. Fig. 7.1.11 lists the primary factors which govern the design strains of composite materials and also shows the relative effect of the environment on laminate structures. To summarize:

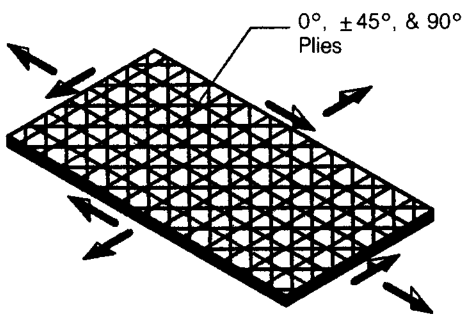
- Under tension loading the governing consideration is notches (holes)
- Compression loading takes over when laminate damage occurs as the result of impact, or notches



(a) Axial loading

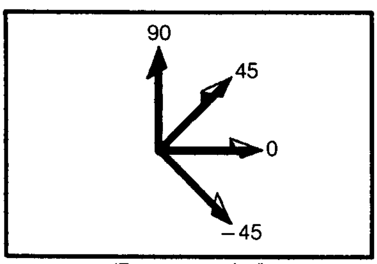


(b) Shear loading



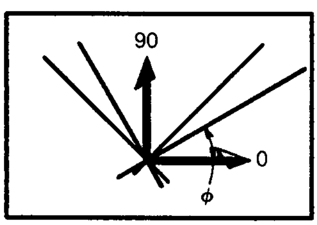
(c) Bi-axial + shear loading

Fig. 7.1.6 Tailoring Ply Orientation to Meet Loading Requirements



(Recommended)

(a) 0, 45, 90 AND −45
Angles only



(Prior approval required for use)

(b) Arbitrary angles

Fig. 7.1.7 Selection of Ply Angles

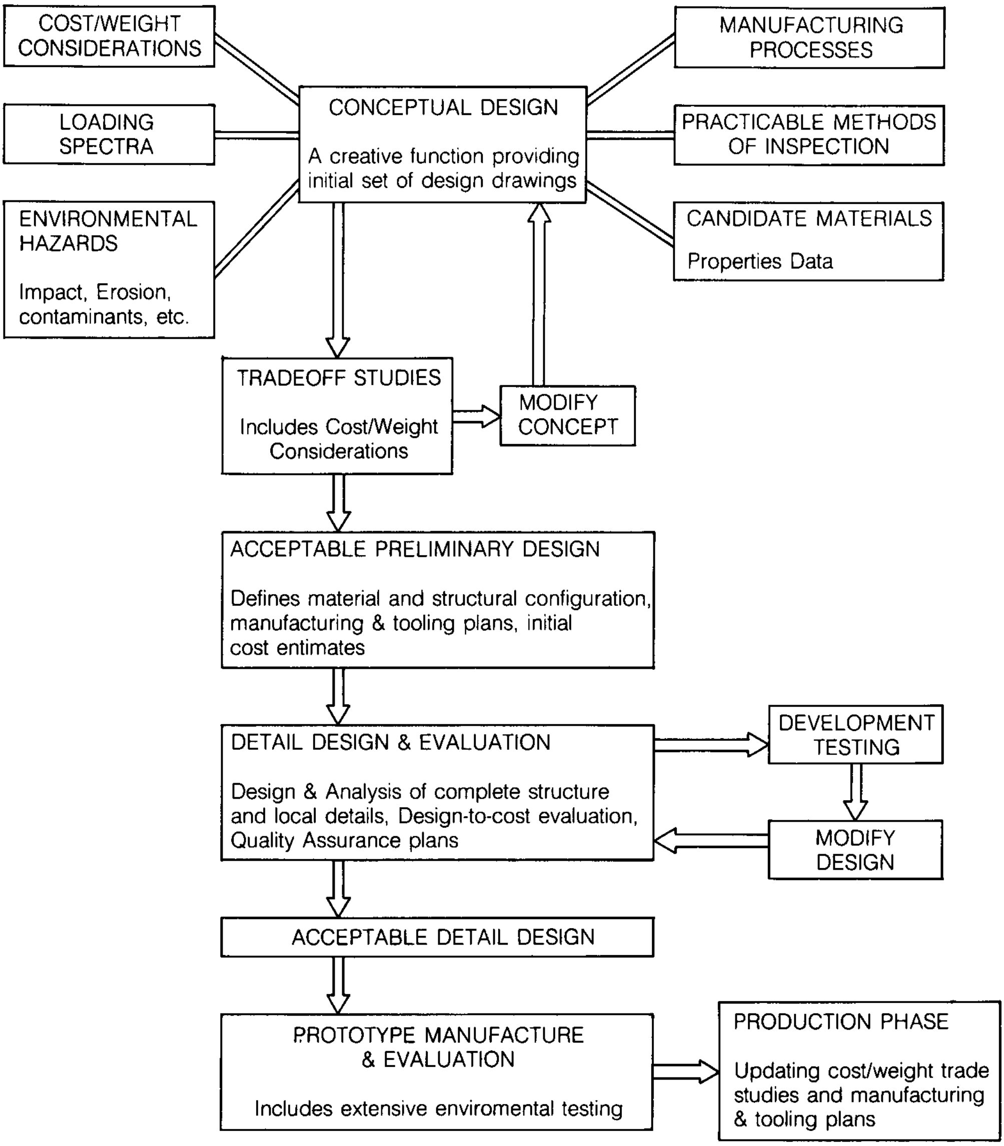


Fig. 7.1.8 Composite Design Methodology

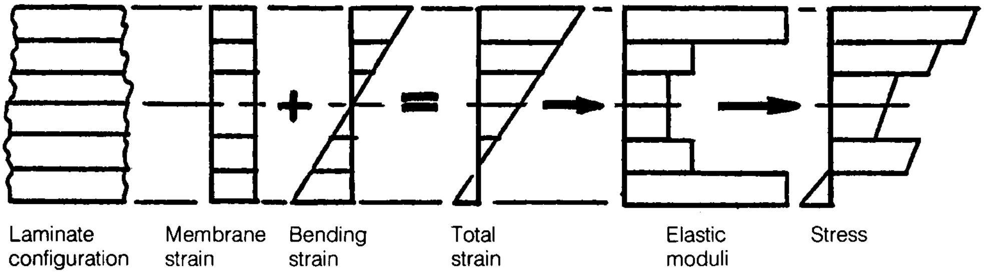
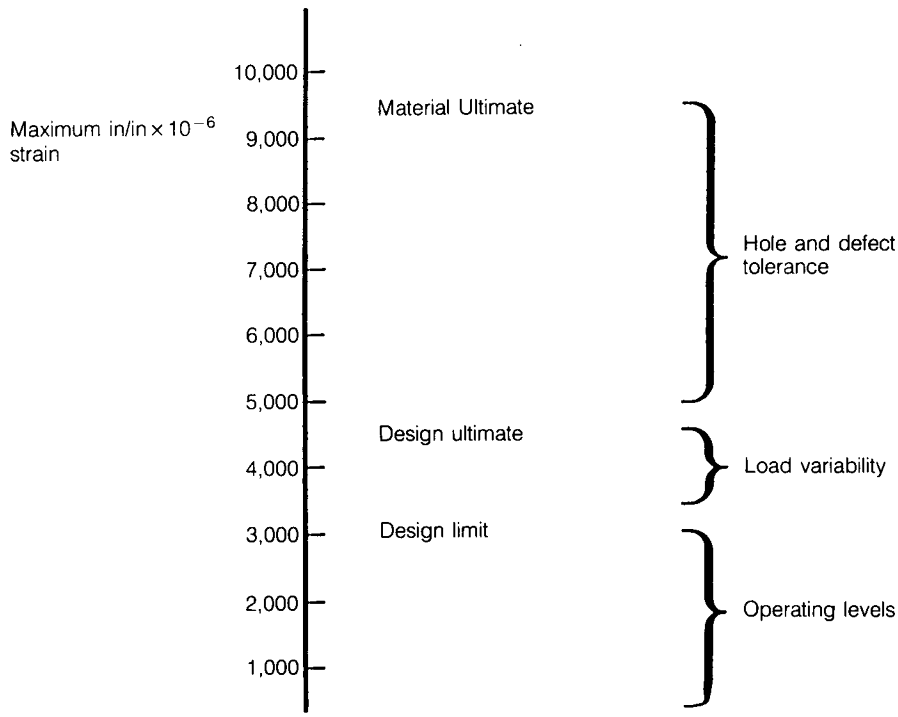
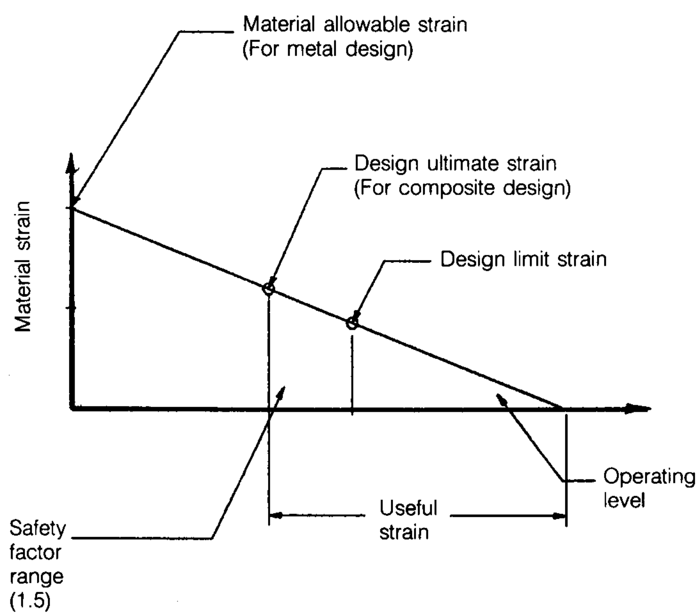


Fig. 7.1.9 Representative Strain, Moduli and Stress Distribution Across the Laminate Thickness

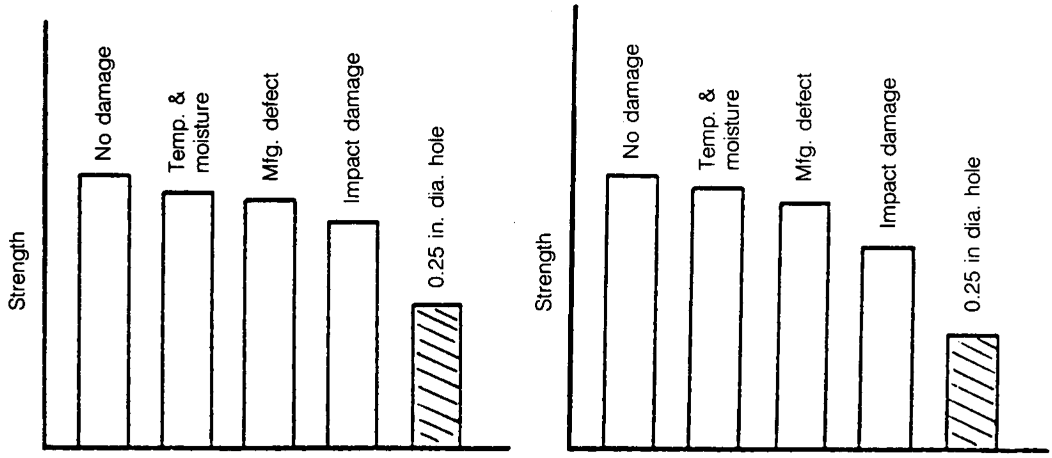


(a) Typical design allowables for thermosets with carbon fibers



(b) Schematic

Fig. 7.1.10 Design Allowable Breakdown



(a) Tension

(b) Compression

Fig. 7.1.11 Factors Affecting Design Strength

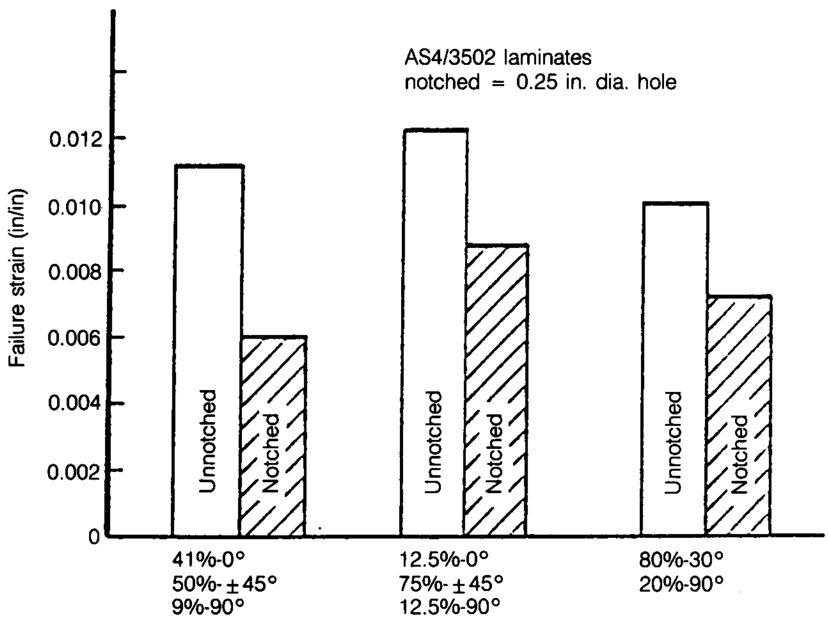


Fig. 7.1.12 Notch Sensitivity Versus Percentage of ±45° Plies

For composite laminates with a given flaw or hole size, strength retention in general increases as the percentage of ±45° plies present in an orientation is increased (see Fig. 7.1.12). However, total load carrying capacity of the laminate generally decrease due to reduced moduli. For example, the stress concentration factor for a drilled hole is more than three times higher in unidirectional carbon/epoxy than in a laminate consisting of only ±45° plies.

Strength and stiffness in a laminate is determined by:

- The brittle nature of composites
- Material form (unitape, woven fabric, 3-D preform, etc.)
- Orientation of plies
- Volume ratio of fibers
- Moisture absorption with elevated temperatures
- Notches or impact damage
- Defects

Design Criteria and Requirements

The design ultimate load (DUL) is the load to be carried by the structure or member without rupture or collapse and is obtained by multiplying the design limit load (DLL) by the factor of safety of 1.5 [see Fig. 7.1.10(b)]. It is important to realize that this factor is intended to cover for accuracy of load, analysis, etc. This factor does not contain any provision for problems of fabrication or processing of composite parts, nor for impact damage, environment, etc. The allowance included in the allowables for material degradation caused by local processing inadequacies is separate from the statistical variations that always occur in the determination of material allowables.

(1) A sound approach to designing composite airframe structures includes the following:

- Analysis methods
- Testing approach
- Material system(s)
- Design concepts (innovative)
- Hybrid construction (e.g., thermosets, thermoplastics, metals, etc.)
- Inspection and repair

In view of the very challenging performance, weight, and cost goals of the airframe project, such influential factors as damage tolerance requirements must be challenged. It is essential that adequate data be available for this and for trade studies in order to produce the best mix of capability and cost.

(2) Selection of ply angle orientation as shown in Fig. 7.1.7:

(a) Case I — Standard angles of 0°, 45°, −45° and 90°:

- These are the basic ply angles commonly used in composite design
- These 4 ply angles satisfy the minimum loading requirements in design and their use is strongly recommended to simplify analysis and manufacturing

(b) Case II — Arbitrary angles:

- Use of limited numbers of non-standard ply angles is allowed only when such use is critical to structural weight reduction and/or a special design
- The number of different ply angles used should be kept to a minimum

(3) Common material thicknesses currently used in composite design:

- Unitape — Approximately 0.005 to 0.0075 inch (0.127 to 0.191 mm)
- Unitape — 0.002 inch (0.0508 mm) (very expensive and only used on space applications)
- Fabric — 0.010 to 0.015 inch (0.25 to 0.381 mm)

7.2 MATERIAL ALLOWABLE DETERMINATION

In the certification process for civil or military aircraft the use of statistically based material properties is required for composite structural analysis. Unidirectional and multidirectional properties of high-strength and high-modulus composites such as carbon/epoxy are shown in Fig. 2.5.5. The unidirectional strength values are not very useful for the design process. Since most structures are subjected to combined loading, it is necessary to orient fibers or individual plies of collimated fibers at specific angles to absorb these loads.

A description and example of the statistical procedures used to derive material design allowables under basic headings of 'A' and 'B' is provided in Ref. 7.12:

(a) 'A' basis — The mechanical property value indicated is the value above which at least 99% of the population of values is expected to fall with a confidence of 95%. This value is used to design a single member whose loading is such that its failure would result in loss of structural integrity

(b) 'B' basis — The mechanical property value indicated is the value above which at least 90% of the population of values is expected to fall with a confidence of 95%. This value is used on redundant or fail-safe structure analysis, where the loads may be safely distributed to other members.

Materials allowables must be obtained by testing (coupon or element tests) as described in Chapter 8.

Statistical analysis of the data has led to the formulation of 'A' and 'B' basis design allowables. 'B' basis design allowables are those mechanical properties that are most commonly used in composite structural analysis.

Composite structures are vulnerable to impact damage and coupon test data indicates that impact damage that is not visible may seriously degrade the compression strength of a laminate. Compression tests should be conducted on elements (later full scale component tests are also required to verify the coupon test data) containing both non-visible and visible impact damage to compare with a laminate containing a given fastener hole diameter (notched). If the compression impact damage is a lower valve than that of notched strength, the compression impact damage strength will be used as the design allowable or maximum design cutoff strain (e.g., 5000 micro in/in for example) for all environmental conditions.

Usually tension allowables are based on a 0.25 inch (6.35 mm) diameter hole, filled or unfilled depending on which is the lower value. Compression allowables are based on a 0.25 inch diameter hole or the damage tolerance requirement. If the hole has a larger diameter or is countersunk, a reduced allowable must be used as shown in Fig. 7.2.1.

Preliminary Design Allowables

Most of the time, a new project will need a group of preliminary allowables to do the initial sizing and analysis before the development of final allowables. Final allowables usually take more than a year to develop during which more than 4000 coupons are tested to generate complete allowable data for use in the certification program. Two methods are described below:

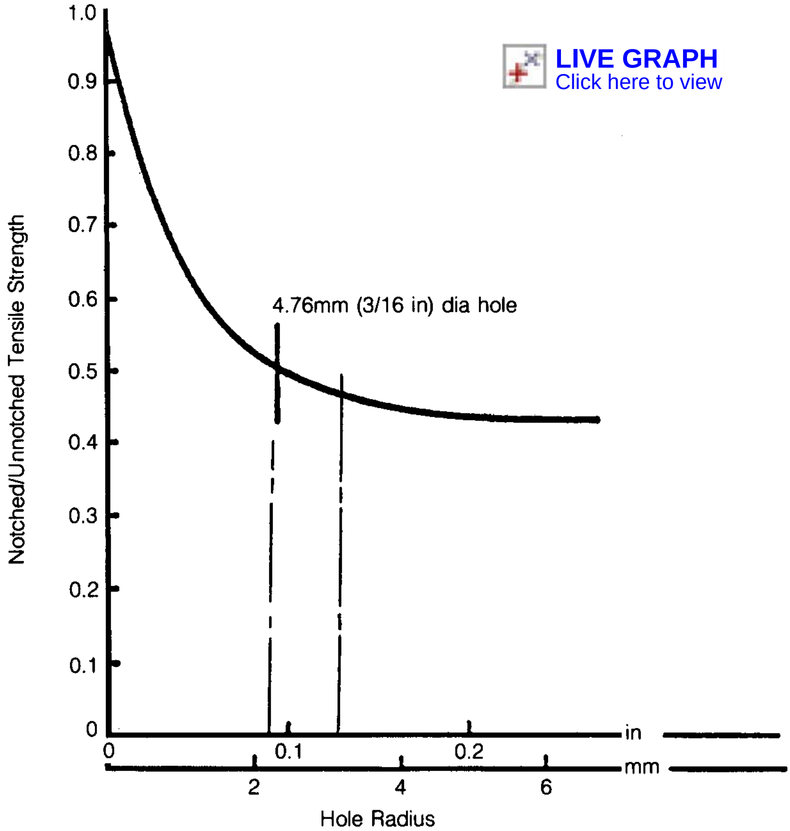
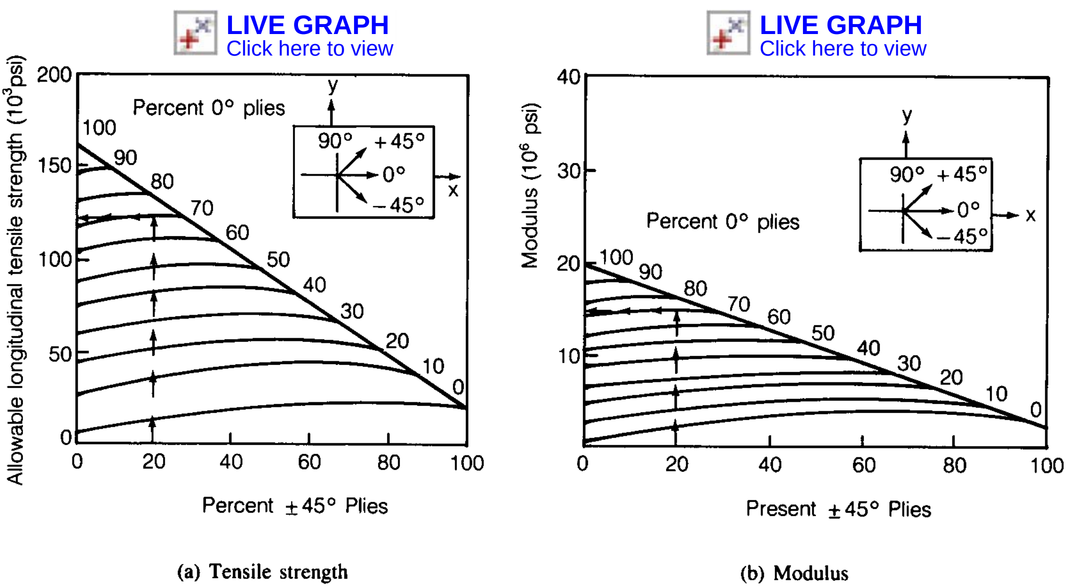


Fig. 7.2.1 Hole Radius Effects on Tensile Strength



(Example only)

Fig. 7.2.2 Carpet Plots of GR/EP Tape Families

(1) Carpet Plot Method (See Coupon Tests in Section 8.3 of Chapter 8):
Carpet plot represent various combinations of symmetric and balanced laminates that contain 0°, 90° and ±45° plies. The ply data may be used to find:
- Properties for a given laminate
- Various laminates that satisfy a particular property requirement

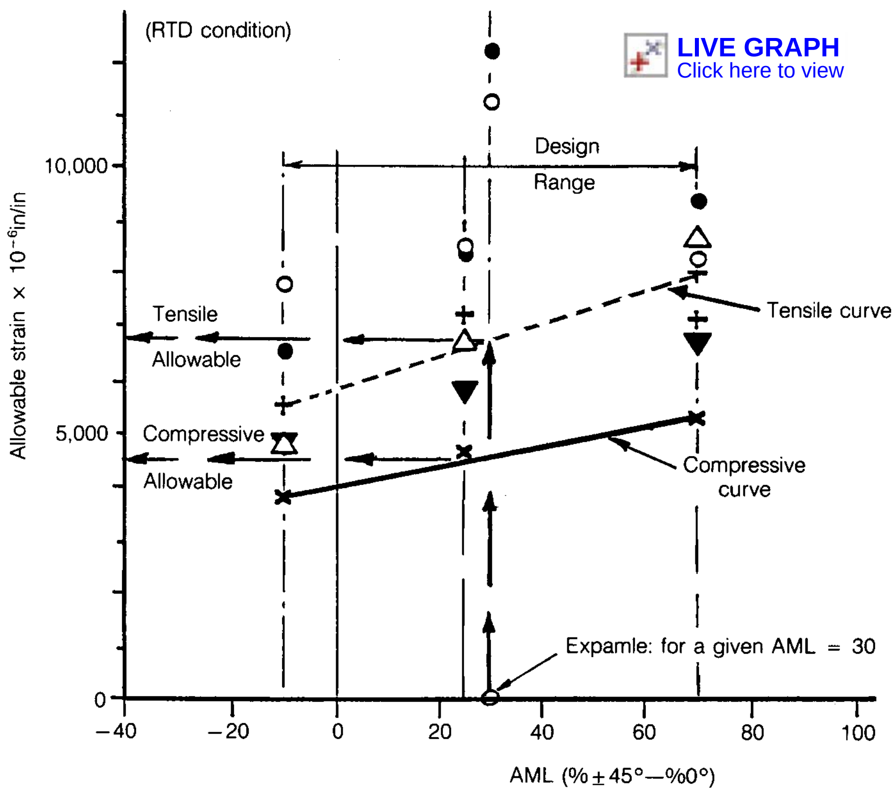
In Fig. 7.2.2 shows two typical carpet plots (modulus and strength values) from a set of design curves for carbon/epoxy unitape (similar plots can be made of woven fabrics). Similar design curves can be found in Ref. 7.13. Such design data curves are only used for the (0/±45/90) family of composite and are only applicable for a particular material system. Consider a carpet plot for each of the following:
- Notched tensile strength
- Compression strength (notched)
- Notched shear strength
- Bearing strength
- Tension, Compression and shear modulus of elasticity
- Poisson's ratio

The notched stresses listed above are gross area and are commonly based on a 0.1785 inch (4.76 mm) or 0.25 inch (6.35 mm) diameter fastener hole with a spacing of 4 times fastener diameter. Each set of the design strain allowable data mentioned above should include following conditions:
- Room temperature dry (RTD)
- Hot/wet
- Cold/dry
- Impact damage

The failure criterion used for predicting strength is the maximum strain criterion with the following assumptions:
- For laminates with fibers in the direction of loading, failure is assumed when laminate strain exceeds the ply-level 0° ply direction failure strain
- For laminates which contain only ±45°/90° plies and which are loaded in tension in the 0° direction, failure is conservatively assumed to occur when the 90° tensile failure strain is exceeded
- When ±45°/90° laminates are loaded in compression, failure is conservatively assumed to occur when the shear strain in the ±45° plies exceeds the ply-level shear strain

(2) AML (Angle Minus Longitudinal) Plots Method (See Coupon Tests in Section 8.3 of Chapter 8):
It has been found in some cases that the failure strain level for stress concentrations varies according to the relative proportions of the various angle plies. Laminates with the highest proportions of 0° plies have the lowest failure strains, and laminates with the highest proportions of ±45° plies have the highest failure strains. Consequently, a current method plots failure strain against 'AML' (Angle Minus Longitudinal, or the % of ±45° plies minus the % of 0° plies; nothing that the term 'angle' refers only to the ±45° plies). The extremes are AML = −100 for all 0° plies and AML = +100 for all ±45° plies.

Laminates of different layups, which provide a range of AML values likely to be used are drilled or impacted and tested to failure. The allowable strain is derived from the test data and plotted as a function of the AML parameter. This method is based on a series of coupon tests (approximately 300 coupons of 16 to 24 ply laminates under room temperature, hot/wet, and cold/dry conditions) including tension tests (open and filled hole) and compression tests including open hole (either a 0.1875 inch or 0.25 inch diameter fastener hole size is commonly selected for the project requirement) and post-impact tests. An AML plot is shown in Fig. 7.2.3. Where holes larger than 0.25 inch diameter exist, further reductions are required locally based on the effect of the largest hole (see Fig. 7.2.1). This method neither accounts for the 90° plies or the stacking sequence, both of which affect the failure strain and it is generally conservative. However, it generally works well for laminates with less than 25% 90° plies. The application of this approach to strength analysis is illustrated in Section 7.6 in this Chapter.



Test data (see Fig. 8.3.22)

○ Open-hole tension
● Filled-hole tension
△ Open-hole compression
▼ Post-impact compression
+ Tensile allowable
× Compressive allowable

Fig. 7.2.3 A Typical AML Plot (Carbon Fiber)

7.3 LAMINATE STRENGTH ANALYSIS

This section contains a brief discussion of the fundamental principles of strain (or stress) at a point in order to form a firm base for the analytical development for composites which follows.

The basic lamination theory is the stress-strain or constitutive relation and failure criteria for the individual laminae. Each lamina consists of a layer of unidirectional fibers or woven cloth, impregnated and fully surrounded by matrix material. The strong, stiff fibers provide the primary load carrying capability while the matrix protects and supports the fibers and transfers the load between them. The terminlogies of the theory of laminated plates are described below:

(a) Micromechanics — The study of the interaction between fiber and matrix in a lamina such that the mechanical behavior of the lamina can be predicted from the known behavior of the constituents:
- Micromechanics establishes the relationship between the properties of the constituents (the fiber and matrix) and those of the unit composite ply
- Complex formulations relating the shape, array and interactions have proven no better than a simple rule-of-mixtures approach based on volume fractions of the constituent
- All approaches require a correction factor to correlate with measured ply level or laminate tests
- All approaches suffer from the problem of measuring the properties of the constituents

  For these reasons, micromechanics is seldom used for aircraft strength or stiffness design. However, it is used for certain physical properties, e.g., density, fiber volume, etc., as will be discussed later.

(b) Macromechanics — The study of the mechanical behavior at any point in a laminate based on the known behavior of the laminae.
- Macromechanics establishes the relationship between the properties of the plies and those of the resultant laminate
- It is based on continuum mechanics which models each ply as homogeneous and orthotropic, ignoring fiber/matrix interface
- Lamination theory is the principle mathematical tool for determining the property relationship between ply and laminate
- The ply properties are determined from ply level or laminae tests

(c) Lamination theory — A mathematical formulation for predicting the macromechanical behavior of a laminate based on an arbitrary assembly of homogeneous orthotropic laminae. Two dimensional theory is most common, and three dimensional theory is most complex.

Laminate strength analysis:
- A ply (or lamina) has five potential modes of failure as shown in Fig. 7.3.1.
- A laminate has additional modes of failure:
  - Delamination and sublaminate buckling
  - Interlaminar shear
  - Interlaminar tension

- A laminate may have failed laminae and continue to carry load
- Strength and stiffness are directional

Poisson's ratio effects:

- Poisson's ratio (ν) for composite materials is directional and varies with relative orientations of the individual plies (see Fog. 7.3.2)
- Composite parts may induce large loads to adjacent structures because of their relatively high Poisson's ratio
- The Poisson's ratio of composites is significantly higher than the usual value of metals
- Special care should be taken at the boundaries between composites and metals where induced stresses due to high Poisson's ratio are greatest

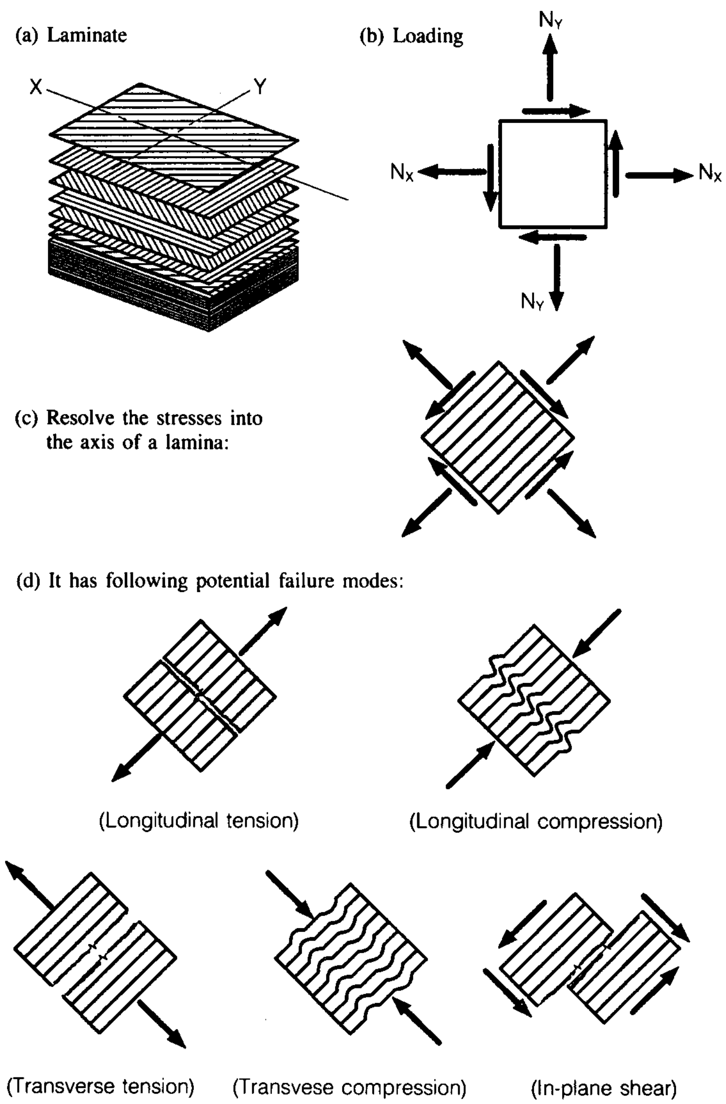


Fig. 7.3.1 Ply Failure Modes

- **The top ply has the most x-direction stress because it is stiffer than the bottom ply in that direction**
- **The x-direction displacements are identical, but**

Without bonding:

$\Delta_y^1 > \Delta_y^2$ (because $\nu_{12} > \nu_{21}$)

With bonding:

— The top ply must get wider (σ_{y_T} = tension)

— And bottom ply must get narrower (σ_{y_B} = compression)

— From equilibrium

$\sigma_{y_T} t_T + \sigma_{y_B} t_B = 0$

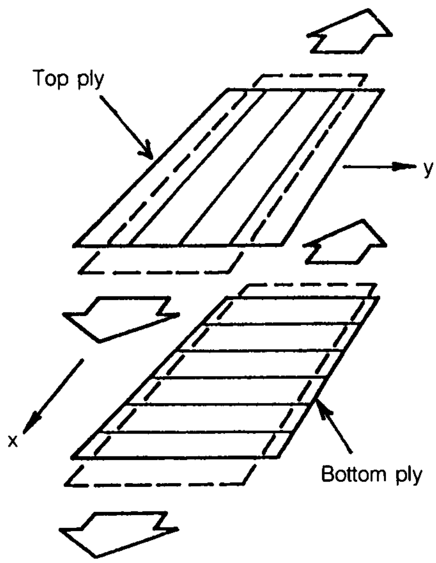
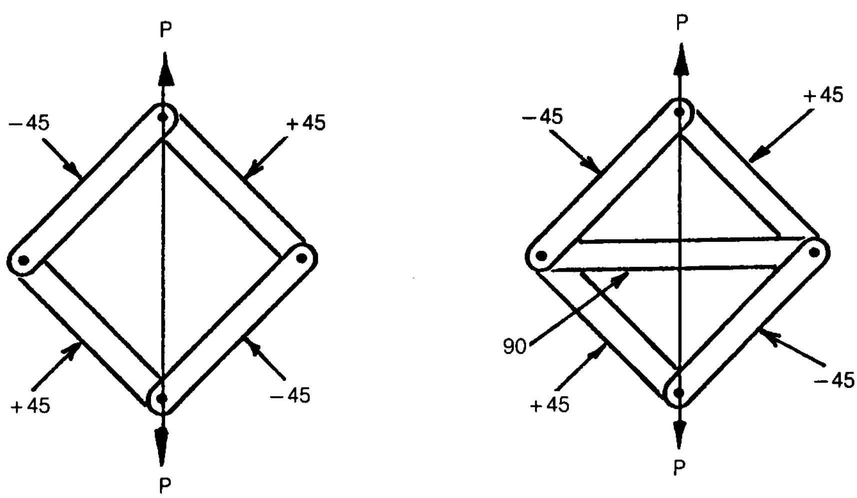


Fig. 7.3.2 Poisson's Ratio Mismatch of Two Plies



±45 Laminate:

Laminate relies on matrix for much of its rigidity

±45, 90 Laminate:

Like a truss, the addition of transverse fibers stiffen the longitudinal direction

Fig. 7.3.3 Lamina Interaction

With the advent of composite materials, there is an opportunity for materials design to be integrated into structural design as an added dimension. A basic understanding of the interaction of fiber, matrix, and fiber-matrix interface in composites (see Fig. 7.3.3) will be a valuable aid to engineers and to materials and structural analysis. The modern science of micromechanics, which is the study of structural material interactions, is of particular importance for composites analysis.

However, the laminate which is required to withstand a number of differing conditions and environments cannot be efficiently designed with micromechanics alone. Techniques of using combinations of 0°, ±45°, and 90° laminae can produce a simple design of relatively good efficiency. The best efficiencies, however, are achieved by computerized techniques (refer to the list of computer programs from "Computer Programs for Structural Analysis" in Ref. 7.17). These computerized techniques are based on both empirical test data and constituent material properties along with micromechanics.

The following study relates the longitudinal and transverse mechanical properties of an unidirectional fiber reinforced composite to the properties of the constituents. The basic equations used in this study to predict the longitudinal modulus and strength of fiber reinforced composites are the parallel element mixture equations. These are based on reasonable assumption and they do not violate theories of elasticity.

$$E_{11} = E_f V_f + E_m(1 - V_f) \tag{7.3.1}$$

$$F_{11} = F_f V_f + F_m(1 - V_f) \text{ (generally in tension)} \tag{7.3.2}$$

where: E_{11} — Modulus of elasticity parallel to fiber length
F_{11} — Material strength parallel to fiber length
E_f — Modulus of elasticity of fiber
F_f — Strength of fiber
V_f — Volume of fiber
E_m — Modulus of elasticity of matrix
F_m — Strength of matrix

The assumptions on which these two equations are based on as follows:

- The fibers are completely sorrounded and wetted by matrix material and accordingly are not allowed to contact one another
- The transfer of load from the matrix to the fiber occurs across the interface surfaces comprising wetted areas
- The strength, size, shape, orientation, and bonding of the fibers are as uniform as possible

The overall Poisson's ravio (ν) of the composite can also be predicted using an equation of similar form.

$$\nu_{12} = \nu_f V_f + \nu_m(1 - V_f) \tag{7.3.3}$$

where ν_{12} — Poisson's ratio of the composite material
ν_f — Poisson's ratio of the fiber
ν_m — Poisson's ratio of the matrix

Predictions of the tensile modulus normal to the fiber direction and the shear modulus in a unidirectional composite are difficult to make because of their sensitivity to voids, and their dependence on accurate knowledge of both the matrix modulus and the details of the fiber-matrix packing. The two equations given below are used to approximate the transverse tensile modulus and shear modulus of unidirectional fiber reinforced composite materials.

$$E_{22} = 1/[(V_f/E_f) + (1 - V_f)/E_m] \quad (7.3.4)$$

$$G_{12} = 1/[(V_f/G_f) + (1 - V_f)/G_m] \quad (7.3.5)$$

where E_m — Modulus of Elasticity of the matrix

G_m — Modulus of Rigidity of the matrix

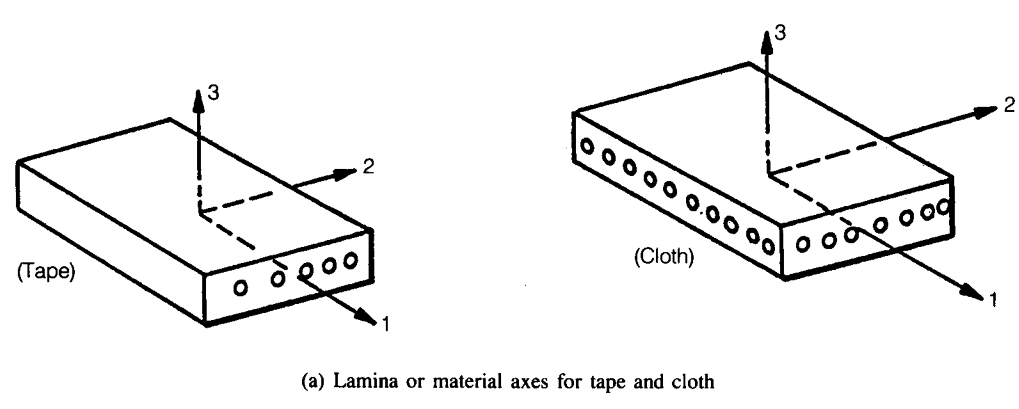
Note that Equations (7.3.1) through (7.3.5) represent but a few of the many micromechanical expressions developed to predict the behavior of fiber reinforced composite materials (for further information, see Ref. 7.3).

Lamination Theory

When the plane stress assumption is valid in lamination theory for a thin plate which is subjected only to inplane or membrane loads that do not cause any instability, the stress-strain behavior become relatively simple. Lamina axes are defined by the numbers 1, 2, and 3 and the laminate is defined by x, y and z, as shown in Fig. 7.3.4.

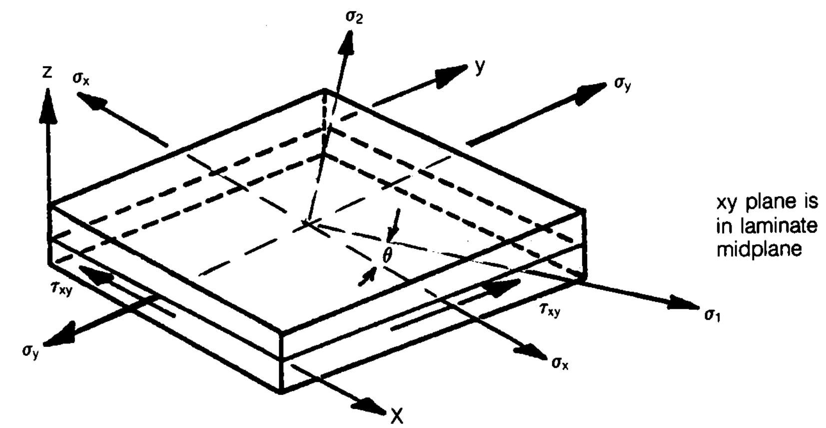
Basic assumptions of lamination theory:

- The structure is restricted to be a thin plate or shell consisting of an arbitrary combination of laminae (plies) cured or consolidated into a single laminated plate
- The assumptions of plate and shell theory hold; i.e., the through plate thickness direct stress (σ_z) is zero and only plane stresses (σ_{xy}) exists
- The two transverse shear stresses (τ_{xz}, τ_{yz}) are neglected to meet the classic thin plate and shell theory [see Fig. 7.3.4(b)]
- The theory is a 'point' analysis in an effectively infinitely large plate and shell, completely ignoring the effects of neighboring edges, stiffeners, holes, cutouts or any other discontinuities
- The loading is assumed to be inplane memberane stress and moment resultants.



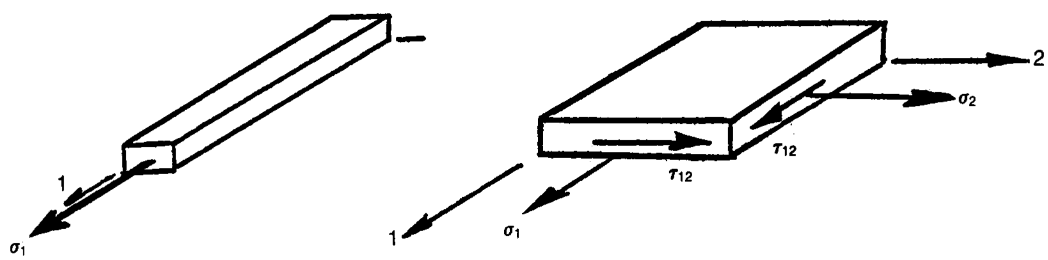
(a) Lamina or material axes for tape and cloth

Fig. 7.3.4 Definition of Composites Coordinates



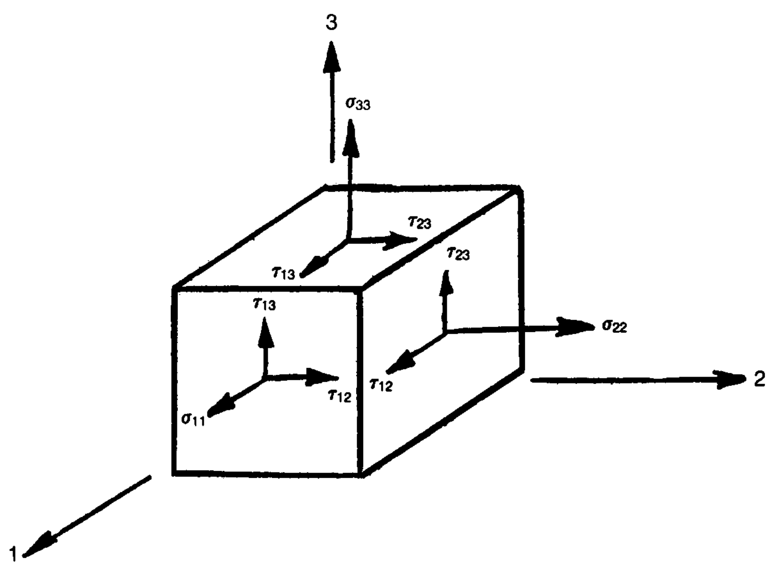
(b) Laminae and laminate coordinates and stresses

Fig. 7.3.4 Definition of Composites Coordinates (cont'd)



(a) One-dimensional

(b) Two-dimensional



(c) Three-dimensional

Fig. 7.3.5 Hooke's Law Theory

Hooke's Law Theory

(1) For a homogeneous isotropic material in a one dimensional stress state, [see Fig. 7.3.5(a)] the Hooke's law relationship is:

$$\sigma = E \times \epsilon \tag{7.3.6}$$

(where the proportionality constant, E, is Young's modulus or the modulus of elasticity and, ϵ, is strain)

(2) For a homogeneous isotropic material in a two dimensional stress state [plane stress, see Fig. 7.3.5(b)], the Hooke's law relationship become:

$$\sigma_1 = (\epsilon_1 + \nu\epsilon_2)\frac{E}{1-\nu^2} \tag{7.3.7}$$

$$\sigma_2 = (\epsilon_2 + \nu\epsilon_1)\frac{E}{1-\nu^2} \tag{7.3.8}$$

$$\tau_{12} = (\gamma_{12})\frac{E}{2(1+\nu)} \tag{7.3.9}$$

or the equations in matrix form are:

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} C_{11} & C_{12} & 0 \\ C_{12} & C_{22} & 0 \\ 0 & 0 & C_{66} \end{bmatrix} \begin{bmatrix} \epsilon_1 \\ \epsilon_2 \\ \gamma_{12} \end{bmatrix} \tag{7.3.10}$$

where,

$$C_{11} = C_{22} = E/(1-\nu^2) \tag{7.3.11}$$

$$C_{21} = C_{12} = \nu E/(1-\nu^2) \tag{7.3.12}$$

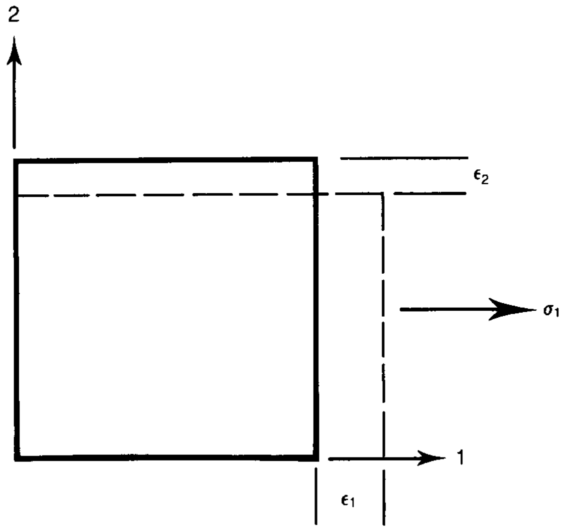
$$C_{66} = E/2(1+\nu) = G \tag{7.3.13}$$

It is evident that two independent elastic constants appear in Eq. (7.3.10). E and ν. The third elastic constant, shear modulus, G. is a function of E and ν. The relationship of stress and strain is shown in Fig. 7.3.6 and

$$G = E/2(1+\nu) \tag{7.3.14}$$

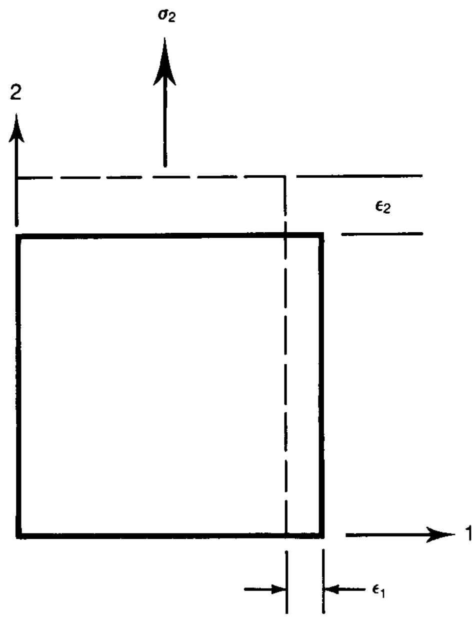
Therefore, for isotropic materials only two elastic constants are necessary to write the Hooke's law relationships for two or three dimensional applications.

(3) The Hooke's law relationships can be generalized for three dimensional anisotropic [see Fig. 7.3.5(c)] materials.
   (a) It requires twenty-one independent elastic constants for the Hooke's law relationships for an anisotropic material in three dimensions.
   (b) For an orthotopic material in three dimensions, only nine independent elastic constants are necessary.
   (c) For the case which is of the most practical interest, an orthotropic material in a two dimensional application, there are four independent elastic constants required to specify a Hooke's law relationship.



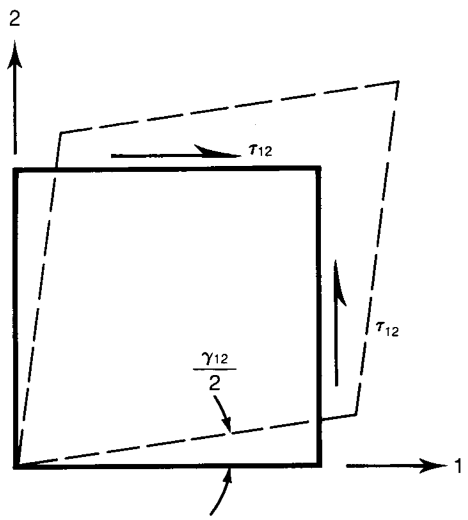
$\epsilon_1 = \sigma_1/E_{11}$
$\epsilon_2 = \nu_{12}\epsilon_1$

(a) Due to stress σ_1



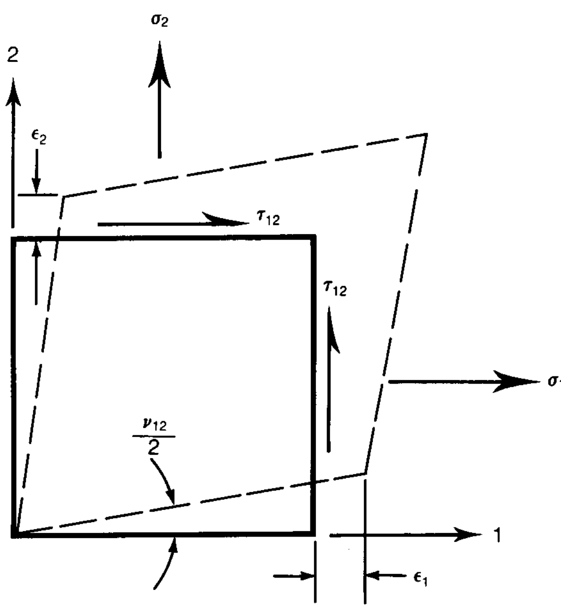
$\epsilon_2 = \sigma_2/E_{22}$
$\epsilon_1 = \nu_{21}\epsilon_2$

(b) Due to stress σ_2



$\frac{\gamma_{12}}{2} = \tau_{12}/2G_{12}$

(c) Due to shear stress τ_{12}



$\epsilon_1 = \sigma_1 \; E_{11} - \nu_{21}\epsilon_2 = \sigma_1/E_{11} - \nu_{21}/E_{22}$
$\epsilon_2 = \sigma_2/E_{22} - \nu_{12}\epsilon_1 = -\nu_{12}\sigma_1/E_{11} + \sigma_{12}/E_{22}$
$\gamma_{12}/2 = \tau_{12}/2G_{12}$

(d) Due to combined stresses

Fig. 7.3.6 Two-dimensional/Stress and Strain Relationship

Case (c) is of particular interest because the individual lamina of a filamentary composite may thus be modeled as an orthotropic material in a state of plane stress.

Generalized Hooke's law now can be written in matrix form (eighty-one elastic constants):

$$\begin{bmatrix} \sigma_{11} \\ \sigma_{22} \\ \sigma_{33} \\ \tau_{23} \\ \tau_{13} \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} C_{11} & C_{12} & C_{13} & C_{14} & C_{15} & C_{16} \\ C_{12} & C_{22} & C_{23} & C_{24} & C_{25} & C_{26} \\ C_{13} & C_{23} & C_{33} & C_{34} & C_{35} & C_{36} \\ C_{14} & C_{24} & C_{34} & C_{44} & C_{45} & C_{46} \\ C_{15} & C_{25} & C_{35} & C_{45} & C_{55} & C_{56} \\ C_{16} & C_{26} & C_{36} & C_{46} & C_{56} & C_{66} \end{bmatrix} \begin{bmatrix} \epsilon_{11} \\ \epsilon_{22} \\ \epsilon_{33} \\ \gamma_{23} \\ \gamma_{13} \\ \gamma_{12} \end{bmatrix} \qquad (7.3.15)$$

(where $[C_{ij}]$ is the stiffness matrix)

For a state of plane stress, the equation above reduces to:

$$\begin{bmatrix} \sigma_{11} \\ \sigma_{22} \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} C_{11} & C_{12} & 0 \\ C_{12} & C_{22} & 0 \\ 0 & 0 & C_{66} \end{bmatrix} \begin{bmatrix} \epsilon_{11} \\ \epsilon_{22} \\ \gamma_{12} \end{bmatrix} \qquad (7.3.16)$$

Where only 4 independent elastic constants exist.

$$C_{11} = \frac{E_{11}}{(1-\nu_{12}\,\nu_{21})} \qquad (7.3.17)$$

$$C_{12} = \nu_{12}\,C_{22} \qquad (7.3.18)$$

$$C_{22} = \frac{E_{22}}{(1-\nu_{12}\,\nu_{21})} \qquad (7.3.19)$$

$$C_{66} = G_{12} \qquad (7.3.20)$$

The following equation is used to transform the elastic constants of an orthotropic material (ply) in two dimensions to laminate axes, x and y (see Fig. 7.3.4). Without going into the derivation, the transformation relationships for the elastic constants for an orthotropic material in a plane stress state and the transformation matrix $[Q_{ij}]$ are as follows:

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{21} & Q_{22} & Q_{26} \\ Q_{61} & Q_{62} & Q_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} \qquad (7.3.21)$$

Where:

$$Q_{11} = C_{11}\cos^4\theta + 2(C_{12} + 2C_{66})\sin^2\theta\cos^2\theta + C_{22}\sin^4\theta \qquad (7.3.22)$$

$$Q_{22} = C_{11}\sin^4\theta + 2(C_{12} + 2c_{66})\sin^2\theta\cos^2\theta + C_{22}\cos^4\theta \qquad (7.3.23)$$

$$Q_{12} = (C_{11} + C_{22} - 4C_{66})\sin^2\theta\cos^2\theta + C_{12}(\sin^4\theta + \cos^4\theta) \qquad (7.3.24)$$

$$Q_{66} = (C_{11} + C_{22} - 2C_{12} - 2C_{66})\sin^2\theta\cos^2\theta + C_{66}(\sin^4\theta + \cos^4\theta) \qquad (7.3.25)$$

$$Q_{16} = (C_{11} - C_{12} - 2C_{66}) \sin\theta \cos^3\theta + (C_{12} - C_{22} + 2C_{66}) \sin^3\theta \cos\theta \quad (7.3.26)$$

$$Q_{26} = (C_{11} - C_{12} - 2C_{66}) \sin^3\theta \cos\theta + (C_{12} - C_{22} + 2C_{66}) \sin\theta \cos^3\theta \quad (7.3.27)$$

Q's elastic constants refer to the laminate x and y coordinates as shown in Fig. 7.3.4

General Constitutive Equation

The constitutive equation for a thin laminated anisotropic plate can be written (not including the term for thermomechanical properties):

$$\left[\frac{N}{M}\right] = \left[\begin{array}{c|c} A & B \\ \hline B & D \end{array}\right] \left[\frac{\epsilon}{\varkappa}\right] \quad (7.3.28)$$

or by the following form:

$$\begin{bmatrix} N_X \\ N_Y \\ N_{XY} \\ \hline M_X \\ M_Y \\ M_{XY} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{21} & A_{22} & A_{26} \\ A_{61} & A_{62} & A_{66} \\ \hline B_{11} & B_{12} & B_{16} \\ B_{21} & B_{22} & B_{26} \\ B_{61} & B_{62} & B_{66} \end{bmatrix} \begin{bmatrix} \epsilon_X \\ \epsilon_Y \\ \epsilon_{XY} \end{bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{21} & B_{22} & B_{26} \\ B_{61} & B_{62} & B_{66} \\ \hline D_{11} & D_{12} & D_{16} \\ D_{21} & D_{22} & D_{26} \\ D_{61} & D_{62} & D_{66} \end{bmatrix} \begin{bmatrix} \varkappa_X \\ \varkappa_Y \\ \varkappa_{XY} \end{bmatrix} \quad (7.3.29)$$

where

$$A_{ij} = \sum_{k=1}^{n} \left(Q_{ij}\right)_k \left(h_k - h_{k-1}\right) \quad (7.3.30)$$

$$B_{ij} = \sum_{k=1}^{n} \left(Q_{ij}\right)_k \left(h_k^2 - h_{k-1}^2\right) /2 \quad (7.3.31)$$

$$D_{ij} = \sum_{k=1}^{n} \left(Q_{ij}\right)_k \left(h_k^3 - h_{k-1}^3\right) /3 \quad (7.3.32)$$

(A_{ij} is the extensional or membrance stiffness. D_{ij} is the flexural or bending stiffness while B_{ij} is responsible for the coupling between membrane and bending behavior; see Fig. 7.3.7 for notations)

Eq. (7.3.28) is the general constitutive equation for laminated composites (plates and shells). In this form, the significant point is that there is coupling between extensional (membrane) deformation and bending deformation caused by the existence of the [B] matrix. In other words, even within the limits of small deflection theory, forced curvature, $[\varkappa]$, within the laminate induces in-plane loads, [N], through this type of coupling. Also, in-plane strains, $[\epsilon]$, would induce curvatures, $[\varkappa]$, in the laminate. This coupling is caused by the neutral axis and the midplane of the laminate not being coincident.

The procedures for simplifying the general equation for practical use are:

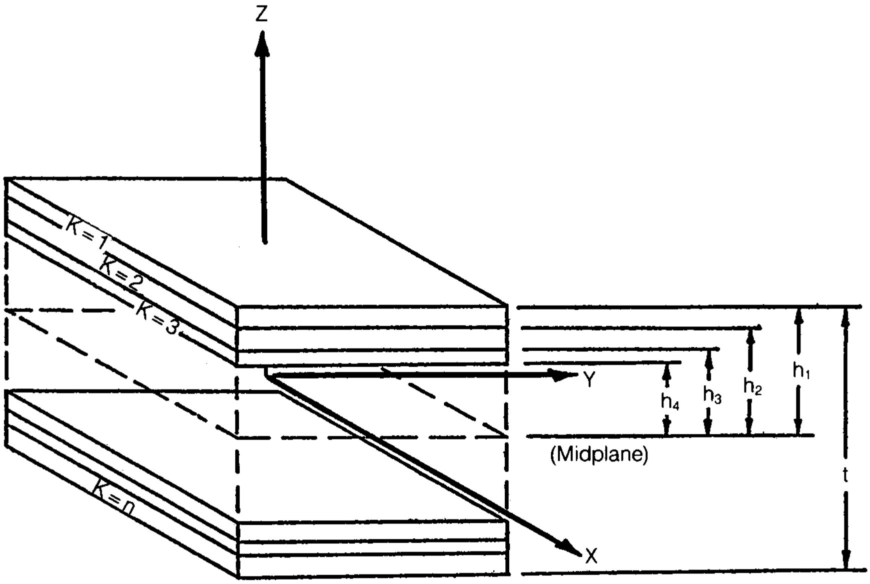


Fig. 7.3.7 Laminate Ply Notation

(a) Eliminate coupling matrix [B] — This can be accomplished by fabricating the laminate symmetrically about the midplane. In other words, an equal number of identical plies (thickness and orientation) are located at the same distance above the midplane as are located below the midplane. With this restriction, the constitutive equation reduces to:

$$[N] = [A]\,[\epsilon] \tag{7.3.33}$$
$$[M] = -[D]\,[\varkappa] \tag{7.3.34}$$

Hence, a laminate which is symmetrically laminated about the midplane is often referred to as "homogeneous" anisotropic.

(b) If the laminate is constructed with equal numbers of pairs of laminae with symmetry about the coordinate (x, y) axes (angle ply laminate, see Fig. 7.3.8), the [A] matrix becomes orthotropic in nature ($A_{16}=A_{26}=0$) as

$$[A] = \begin{bmatrix} A_{11} & A_{12} & 0 \\ A_{12} & A_{22} & 0 \\ 0 & 0 & A_{66} \end{bmatrix} \tag{7.3.35}$$

The [D] matrix remains fully populated and anisotropic in nature.

(c) If the laminate is constructed with equal numbers of pairs of laminae at angles of 0° and 90° (cross-ply laminate, see Fig. 7.3.9) to the x-y axes, the [D] matrix will become orthotropic in nature.

Finally, the practical equation may be written

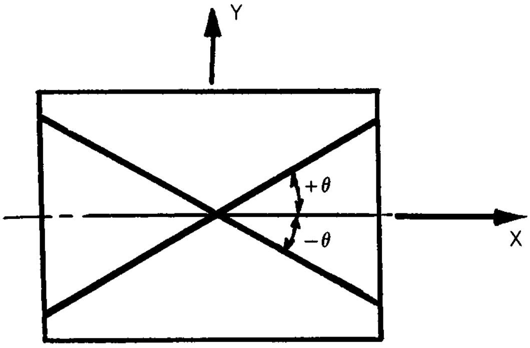


Fig. 7.3.8 Symmetric-ply Laminate

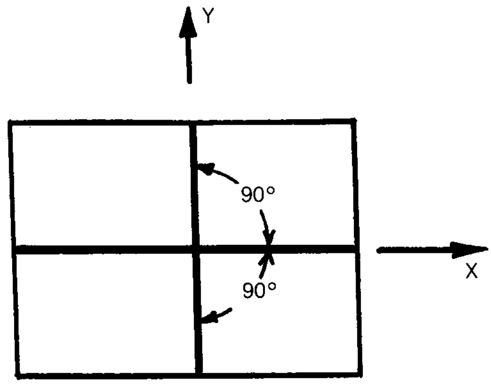


Fig. 7.3.9 Cross-ply Laminate

$$\begin{bmatrix} N_x \\ N_y \\ N_{xy} \end{bmatrix} = [A] \begin{bmatrix} \epsilon_x \\ e_y \\ \gamma_{xy} \end{bmatrix} \tag{7.3.36}$$

Dividing both sides by the total laminate thickness, t, yields

$$\begin{bmatrix} \bar{\sigma}_x \\ \bar{\sigma}_y \\ \bar{\tau}_{xy} \end{bmatrix} = \frac{1}{t}[A] \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} = [\bar{A}] \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} \tag{7.3.37}$$

This practical equation can be used for the following reasons:

- The majority of the laminates which will be utilized for design will necessarily be symmetrical about the midplane or nearly so because of the warpage which will occur during the fabrication process when symmetry does not exist.
- The majority of the applications for advanced composites in the past have been largely confined to structures where membrane loading exists due to the relatively low transverse shear strength of the laminates.

From Eq. (7.3.37), the σ's are the average laminate stresses. The [A] matrix is the laminate stiffness matrix just as the [C] and [Q] matrices, were the lamina stiffness matrices. The equation may be inverted and yield:

$$\begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} = [\bar{A}^*] \begin{bmatrix} \bar{\sigma}_x \\ \bar{\sigma}_y \\ \bar{\tau}_{xy} \end{bmatrix} \tag{7.3.38}$$

(where the laminate compliance matrix is given by $[\bar{A}^*] = [\bar{A}]^{-1}$)

The gross or average laminate elastic moduli may be obtained from the components of the laminate compliance matrix $[\bar{A}^*.]$ The gross laminate elastic constants are given by the following:

$$E_{xx} = \frac{1}{\bar{A}^*_{11}} = \frac{A_{11}A_{22} - A^2_{12}}{A_{22}\,t} \qquad (7.3.39)$$

$$E_{yy} = \frac{1}{\bar{A}^*_{22}} = \frac{A_{11}A_{22} - A^2_{12}}{A_{11}\,t} \qquad (7.3.40)$$

$$G_{xy} = \frac{1}{\bar{A}^*_{66}} = \frac{A_{66}}{t} \qquad (7.3.41)$$

$$\nu_{xy} = \frac{-\bar{A}^*_{12}}{\bar{A}^*_{11}} = \frac{A_{12}}{A_{22}} \qquad (7.3.42)$$

$$\nu_{yx} = \frac{-\bar{A}^*_{12}}{\bar{A}^*_{11}} = \frac{A_{12}}{A_{11}} \qquad (7.3.43)$$

By referring to Fig. 7.3.4(b) and Fig. 7.3.10 and using the right-hand-rule sign convention, the laminate stress resultants and stress couples may be determined as follows:

$$N_X = \int_{-t/2}^{t/2} \sigma_x \, dz \qquad (7.3.44)$$

$$N_y = \int_{-t/2}^{t/2} \sigma_y \, dz \qquad (7.3.45)$$

$$N_{xy} = \int_{-t/2}^{t/2} \tau_{xy} \, dz \qquad (7.3.46)$$

$$M_x = \int_{-t/2}^{t/2} Z \, \sigma_x \, dz \qquad (7.3.47)$$

$$M_y = \int_{-t/2}^{t/2} Z \, \sigma_y \, dz \qquad (7.3.48)$$

$$M_{xy} = \int_{-t/2}^{t/2} Z \, \tau_{xy} \, dz \qquad (7.3.49)$$

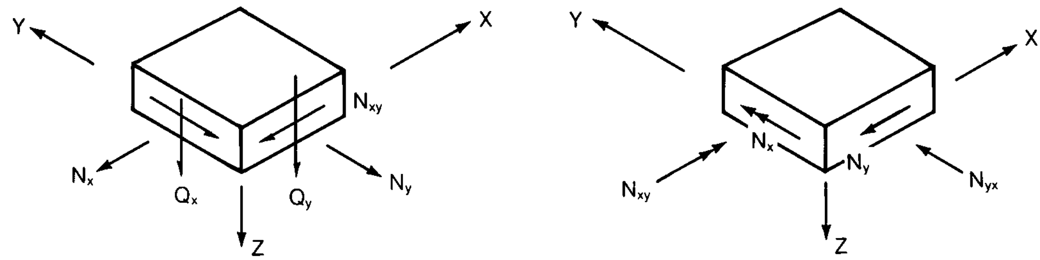


Fig. 7.3.10 Laminate Load Sign Convention

The macromechanics approach to fibrous composites is to model the individual ply or lamina as a homogeneous orthotropic medium subjected to plane stress. Laminate properties (stiffness) have been very successfully predicted from the individual ply properties and good agreement has been obtained with empirical data. Therefore, this technique of utilizing the individual lamina or ply properties to predict laminate properties seems to be the most advantageous approach to the mechanics of laminated composites.

Examples:

Example I:

Material Properties (Lamina or ply):

$$E_{11} = 18.2 \times 10^6 \text{ psi}$$
$$E_{22} = 1.82 \times 10^6 \text{ psi}$$
$$G_{12} = 1 \times 10^6 \text{ psi}$$
$$\nu_{12} = 0.30,\ \nu_{21} = 0.03,\ \nu_{21}E_{11} = \nu_{12}E_{22}$$

(a) Lamina Stiffness Matrix (from Eq. (7.3.16) through Eq. (7.3.20))

$$[C] = \begin{bmatrix} C_{11} & C_{12} & 0 \\ C_{12} & C_{22} & 0 \\ 0 & 0 & C_{66} \end{bmatrix}$$

$$C_{11} = \frac{E_{11}}{1 - \nu_{12}\nu_{21}} = \frac{18.2 \times 10^6}{1-0.3\times0.03} = 20 \times 10^6$$

$$C_{12} = \frac{\nu_{12}E_{22}}{1 - \nu_{12}\nu_{21}} = \frac{0.3 \times 1.82 \times 10^6}{1 - 0.3 \times 0.03} = 0.6 \times 10^6$$

$$C_{22} = \frac{E_{22}}{1 - \nu_{12}\nu_{21}} = \frac{1.82 \times 10^6}{1 - 0.3 \times 0.3} = 2 \times 10^6$$

$$C_{66} = G_{12} = 1 \times 10^6$$

(b) Transform stiffness matrix from lamina to laminate axes (lamina axes 1 & 2 to laminate axes x & y coordinates):

$$[Q] = \begin{bmatrix} Q_{11} & Q_{12} & Q_{16} \\ Q_{12} & Q_{22} & Q_{26} \\ Q_{16} & Q_{26} & Q_{66} \end{bmatrix}$$

Let $\cos\theta = m$ and $\sin\theta = n$

Now from Eq. 7.3.22 through Eq. 7.3.27:

$$Q_{11} = C_{11}m^4 + 2\,(C_{12} + 2C_{66})\,n^2m^2 + C_{22}n^4$$

$$Q_{22} = C_{11}n^4 + 2(C_{12} + 2C_{66})\,n^2m^2 + C_{22}m^4$$

$$Q_{12} = (C_{11} + C_{22} - 4C_{66})\,n^2m^2 + C_{12}\,(n^4 + m^4)$$

$$Q_{66} = (C_{11} + C_{22} - 2C_{12} - 2C_{66})\,n^2m^2 + C_{66}\,(n^4 + m^4)$$

$$Q_{16} = (C_{11} - C_{12} - 2C_{66})\,n\,m^3 + (C_{12} - C_{22} + 2C_{66})\,n^3\,m$$

$$Q_{26} = (C_{11} - C_{12} - 2C_{66})\,n^3\,m + (C_{12} - C_{22} + 2C_{66})n\,m^3$$

For 0° Ply $\quad m = 1;\ n = 0$

$$[Q] = \begin{bmatrix} 20 & .6 & 0 \\ .6 & 2 & 0 \\ 0 & 0 & 1 \end{bmatrix} \times 10^6$$

For 90° Ply $\quad m = 0;\ n = 1$

$$[Q] = \begin{bmatrix} 2 & .6 & 0 \\ .6 & 20 & 0 \\ 0 & 0 & 1 \end{bmatrix} \times 10^6$$

For 45° Ply $\quad m = 1/\sqrt{2};\ n = 1/\sqrt{2}$

$$[Q] = \begin{bmatrix} 6.8 & 4.8 & 4.5 \\ 4.8 & 6.8 & 4.5 \\ 4.5 & 4.5 & 5.2 \end{bmatrix} \times 10^6$$

For $-45°$ Ply $\qquad m = 1/\sqrt{2};\ n = -1/\sqrt{2}$

$$[Q] = \begin{bmatrix} 6.8 & 4.8 & -4.5 \\ 4.8 & 6.8 & -4.5 \\ -4.5 & -4.5 & 5.2 \end{bmatrix} \times 10^6$$

Example II

(A) Consider a 4 ply laminate (see Fig. 7.3.11):

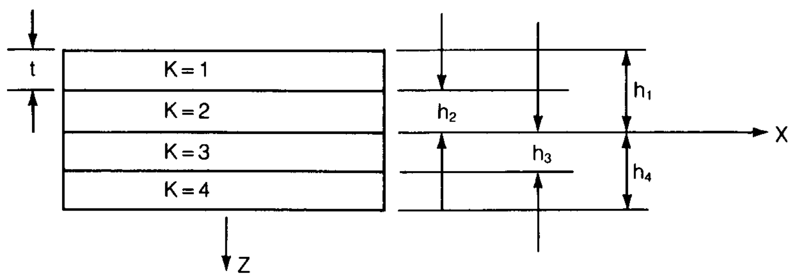


Fig. 7.3.11 Laminate Geemetry (4 Plies)

Thickness per ply $= t$

$h_1 = -2t$

$h_2 = -t$

$h_3 = t$

$h_4 = 2t$

Membrance stiffness [(Eq. (7.3.30)]:

$$A_{ij} = Q_{ij}^{(1)}\left(-t - (-2t)\right) + Q_{ij}^{(2)}\left(0 - (-t)\right) + Q_{ij}^{(3)}(t - 0) + Q_{ij}^{(4)}(2t - t)$$

$$= Q_{ij}^{(1)}t + Q_{ij}^{(2)}t + Q_{ij}^{(3)}t + Q_{ij}^{(4)}t$$

Coupling (Eq. 7.3.31]:

$$B_{ij} = \frac{1}{2}\left[Q_{ij}^{(1)}\left((-t)^2 - (-2t)^2\right) + Q_{ij}^{(2)}\left(0 - (-t)^2\right) + Q_{ij}^{(3)}(t^2 - 0) + Q_{ij}^{(4)}\left((2t)^2 - t^2\right)\right]$$

$$= \frac{1}{2}\left[Q_{ij}^{(1)}(t^2 - 4t^2) + Q_{ij}^{(2)}(-t^2) + Q_{ij}^{(3)}(t^2) + Q_{ij}^{(4)}(4t^2 - t^2)\right]$$

$$= Q_{ij}^{(1)}\left(-\frac{3}{2}t^2\right) + Q_{ij}^{(2)}\left(-\frac{t^2}{2}\right) + Q_{ij}^{(3)}\frac{t^2}{2} + Q_{ij}^{(4)}\left(\frac{3t^2}{2}\right)$$

Flexural stiffness (Eq. (7.3.32)]:

$$D_{ij} = \frac{1}{3}\Big[Q_{ij}^{(1)}\left((-t)^3 - (-2t)^3\right) + Q_{ij}^{(2)}\left(0 - (-t)^3\right) + Q_{ij}^{(3)}\left(t^3 - 0\right)$$

$$+ Q_{ij}^{(4)}\left((2t)^3 - t^3\right)\Big]$$

$$= \left[\frac{1}{3}\, Q_{ij}^{(1)}\,(-t^3 + 8t^3) + Q_{ij}^{(2)}\,(t^3) + Q_{ij}^{(3)}\,(t^3) + Q_{ij}^{(4)}\,(8t^3 - t^3)\right]$$

$$= Q_{ij}^{(1)}\,\frac{7}{3}t^3 + Q_{ij}^{(2)}\,\frac{1}{3}t^3 + Q_{ij}^{(3)}\,\frac{1}{3}t^3 + Q_{ij}^{(4)}\,\frac{7}{3}t^3$$

Values of A_{ij}, B_{ij} and D_{ij} are tabulated below:

	$Q_{ij}^{(1)}$	$Q_{ij}^{(2)}$	$Q_{ij}^{(3)}$	$Q_{ij}^{(4)}$
A_{ij}	t	t	t	t
B_{ij}	$-3/2\ t^2$	$-1/2\ t^2$	$1/2\ t^2$	$3/2\ t^2$
D_{ij}	$7/3\ t^3$	$1/3\ t^3$	$1/3\ t^3$	$7/3\ t^3$

Matrix [A] is independent of stacking sequence. 0/90/90/0, 90/0/90/0 and 0/0/90/90 have identical Matrix [A].

Matrix [B] is only zero if laminate is balanced. 0/90/90/0, $\theta/-\theta/-\theta/\theta$. $\theta/-\theta/\theta/-\theta$ will not have zero Matrix [B]; e.g., 45/−45/45/−45.

$$B_{16} \text{ and } B_{26} = \left[4.5\left(-\frac{3t^2}{2}\right) - 4.5\left(-\frac{t^2}{2}\right) + 4.5\left(\frac{t^2}{2}\right) - 4.5\left(\frac{3t^2}{2}\right)\right] 10^6$$

$$= [9t^2]\ 10^6$$

Matrix [D] is dependent on stacking sequence.

Note: D_{16} and D_{26} are zero for $+\theta/-\theta/+\theta/-\theta$ but not for $+\theta/-\theta/-\theta/+\theta$; e.g., +45/−45/+45/−45 and D_{16} value is:

$$D_{16} = \left[4.5\left(\frac{7t^3}{3}\right) - 4.5\left(\frac{t^3}{3}\right) + 4.5\left(\frac{t^3}{3}\right) - 4.5\left(\frac{7t^3}{3}\right)\right] 10^6 = 0 = D_{26}$$

Under layup of +45/−45/−45/+45 and D_{16} value is:

$$D_{16} = \left[4.5\left(\frac{7t^3}{3}\right) - 4.5\left(\frac{t^3}{3}\right) - 4.5\left(\frac{t^3}{3}\right) + 4.5\left(\frac{7t^3}{3}\right)\right] 10^6$$

$$= [18t^3]\ 10^6 = D_{26}$$

(B) Consider an 8 ply laminate.

	$Q_{ij}^{(1)}$	$Q_{ij}^{(2)}$	$Q_{ij}^{(3)}$	$Q_{ij}^{(4)}$	$Q_{ij}^{(5)}$	$Q_{ij}^{(6)}$	$Q_{ij}^{(7)}$	$Q_{ij}^{(8)}$
A_{ij}	t	t	t	t	t	t	t	t
B_{ij}	$-7/2\ t^2$	$-5/2\ t^2$	$-3/2\ t^2$	$-1/2\ t^2$	$1/2\ t^2$	$3/2\ t^2$	$5/2\ t^2$	$7/2\ t^2$
D_{ij}	$37/3\ t^3$	$19/3\ t^3$	$7/3\ t^3$	$1/3\ t^3$	$1/3\ t^3$	$7/3\ t^3$	$19/3\ t^3$	$37/3\ t^3$

Matrix [A] [B] and [D] are summarized below:

Layup sequence	$[90/0/45/-45]_s$	$[90/-45/0/45]_s$	$[0/45/-45/90]_s$
Matrix [A] =	$\begin{bmatrix} 71.2 & 21.6 & 0 \\ 21.6 & 71.2 & 0 \\ 0 & 0 & 24.8 \end{bmatrix} 10^6 t$	$\begin{bmatrix} 71.2 & 21.6 & 0 \\ 21.6 & 71.2 & 0 \\ 0 & 0 & 24.8 \end{bmatrix} 10^6 t$	$\begin{bmatrix} 71.2 & 21.6 & 0 \\ 21.6 & 71.2 & 0 \\ 0 & 0 & 24.8 \end{bmatrix} 10^6 t$
Matrix [B] =	$\begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} 10^6 t^2$	$\begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} 10^6 t^2$	$\begin{bmatrix} 0 & 0 & 0 \\ 0 & 0 & 0 \\ 0 & 0 & 0 \end{bmatrix} 10^6 t^2$
Matrix [D] =	$\begin{bmatrix} 339 & 48 & 18 \\ 48 & 555 & 18 \\ 18 & 18 & 65 \end{bmatrix} 10^6 t^3$	$\begin{bmatrix} 233 & 81 & -54 \\ 81 & 593 & -54 \\ -54 & -54 & 98 \end{bmatrix} 10^6 t^3$	$\begin{bmatrix} 613 & 99 & 36 \\ 99 & 301 & 36 \\ 36 & 36 & 115 \end{bmatrix} 10^6 t^3$

Design Failure Criteria

Determining whether a design adequately meets design criteria is more difficult with composites than with conventional metal materials because failure criteria are more complex in composites, which can be delaminate or have either matrix or fiber failure. Therefore, most engineers use conservative approaches which needlessly limit where composites can be applied.

Since composite failure modes often operate concurrently, interactively, sequentially with the different modes of failure of a unidirectional fiber composite laminate. The most

- Establish low design strain level from testing
- No matrix failure at limit load
- No fiber failure at ultimate load

Fig. 7.3.12 shows a reasonable set of criteria that satisfy the desire to deal separately with the different modes of failure of a unidirectional fiber composite laminate. The most commonly used are the maximum strain, the maximum stress, and the quadratic criteria which is an empirical description of the failure of a composite component subjected to complex states of stress or strains.

(a) Maximum strain theory:

- No interaction between strains
- Failure occurs whenever:

  $\epsilon_1 = \epsilon_L^{UT}$ or ϵ_L^{UC}

  $\epsilon_2 = \epsilon_T^{UT}$ or ϵ_T^{UC}

  $\gamma_{12} = \gamma_{LT}^{SU}$

  where: UT — Ultimate tension strain or stress

  UC — Ultimate compression strain or stress

  SU — Ultimate shear strain or stress

  Y — Yield stress or strain

  T — Longitudinal direction

  L — Transverse direction

  LT — Long transverse

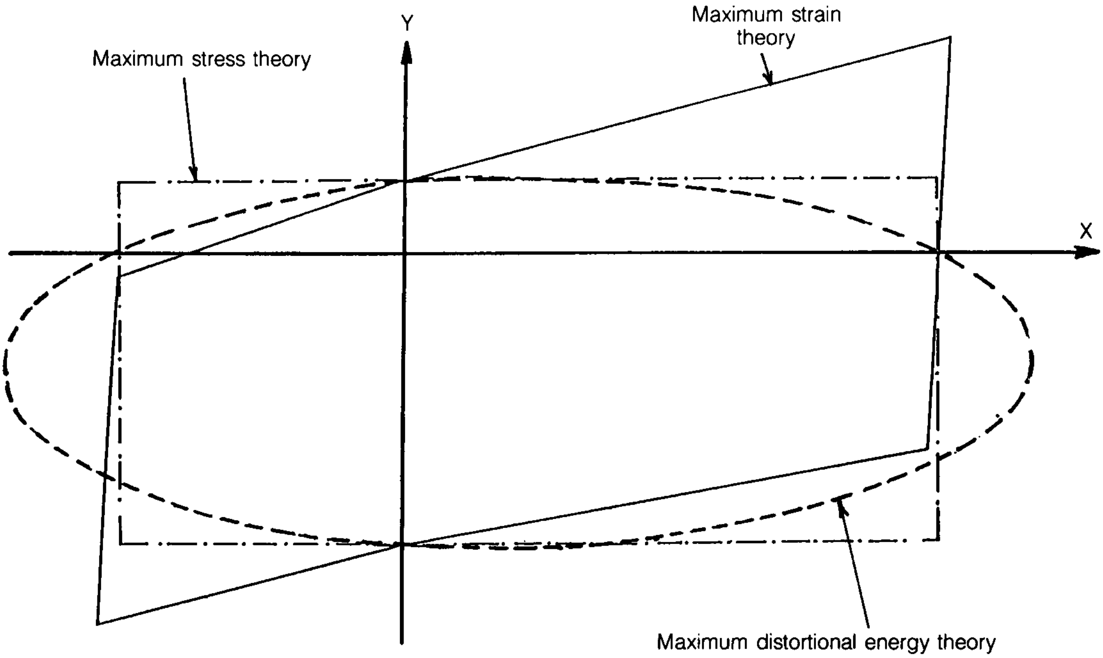


Fig. 7.3.12 Design Failure Strength Envelope Comparison (Ref. 7.16)

(b) Maximum stress theory:
- No interaction between stresses
- Failure occurs whenever:
  $F_1 = F_L^{UT}$ or F_L^{UC}
  $F_2 = F_T^{UT}$ or F_T^{UC}
  $F_{12} = F_{LT}^{SU}$

(c) Hill's (Azzi-Tsai) maximum distortional energy theory:
- Based on metallic yielding theory, interaction between stresses
- Elliptical curve-fit
- Yielding occurs whenever:
  $$(F^2_1 - F_1F_2)/(F^Y{}_L)^2 + F^2{}_2/(F^Y{}_T)^2 + F^2{}_{12}/(F^{SU}{}_{LT})^2 = 1$$

(d) Tsai-Wu strength tensor theory:
- Tensorial so that it is subject to transformation
- Interaction between stresses
- Failure occurs whenever:
  $K_iF_i + K_{ij}\ F_{ij} = 1\ (i,j = 1, 2, 6)$
  Where K_i, K_{ij} — strength tensors (inverse) that require off-axis tensile tests to evaluate

Free Edge Effects

Simple tension and compression tests of crossplied laminates have exhibited significantly less strength than predicted by lamination theory. Frequently, the failure mode in these cases has been a delamination starting at the laminate edges and often propagating across the width. For the same amount of the various plies in a laminate, the failure stresses have varied quite strongly, depending on the stacking sequence or the order in which the plies were laid up. These failures resulted from interlaminar stresses exceeding the matrix strength and edge cracks opening up between the plies at the edges. Within laminate thickness of any edge, whether straight or on a hole or cutout circumference, the interlaminar shear and normal stresses ignored by lamination theory can become significant.

These stresses arise because orthotropic lamina stacked with their material axes at different angles have dissimilar inplane Poisson's ratios, in relation to the laminate axes. This produces different inplane stresses in each lamina whose moment is balanced by interlaminar stresses.

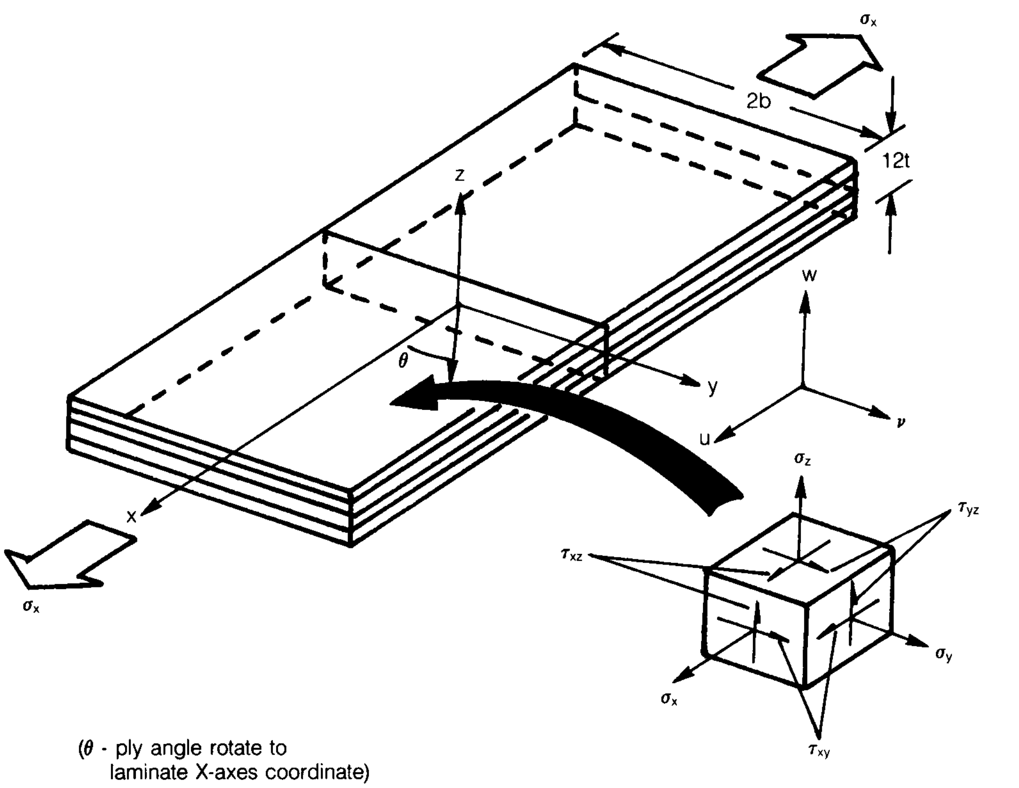
The stacking sequence can cause interlaminar normal stresses to occur at the free edge of the laminate as shown in Fig. 7.3.13. Interlaminar tension stresses can cause delamination under both static and cyclic loading. The sign (tension or compression) of the normal stress depends both on the sign of the laminate inplane loading and the stacking sequence. A given ply set can be stacked in such a way that maximum or minimum tension or compression σ_z can be obtained. In classical lamination theory, no account is taken of interlaminar stress such as σ_z, σ_{xz} and σ_{yz}. Accordingly, classical lamination theory is incapable of providing predictions of some of the stresses that actually cause failure in a composite material. Interlaminar stresses are one of the failure mechanisms uniquely characteristic to composite materials. Moreover, classic lamination theory implies values of τ_{xy} where it cannot possible exist, namely at the edge of a laminate. Physical grounds can be used to establish that:

- At the edges of a laminate, (or hole), the interlaminar shearing stress is very high (perhaps even singular) and would therefore cause the debonding that has been observed in these areas
- Layer stacking sequence changes produce differences in the tensile strength of a laminate even though the orientations of each layer do not change (in classical lamination theory, such changes would not effect stiffness). Interlaminar normal stress (σ_z) changes near the laminate boundaries are believed to provide the answer to such strength differences

The effect of stacking sequence on the interlaminar stress is shown in Fig. 7.3.14.

It is therefore important to consider the stacking sequence in selected orientations to minimize interlaminar stresses. To prevent free edge effects:

- It is good practice to minimize the angle between adjacent tape plies
- The ply stacking sequence should be as homogeneous as possible in order to reduce the effects of interlaminar stress
- Drop-off plies of the same orientation should be dispersed as uniformly as possible
- Keep to a minimum the grouping of plies of the same orientation
- Unitapes are expected to be more sensitive to edge effects than cloth plies



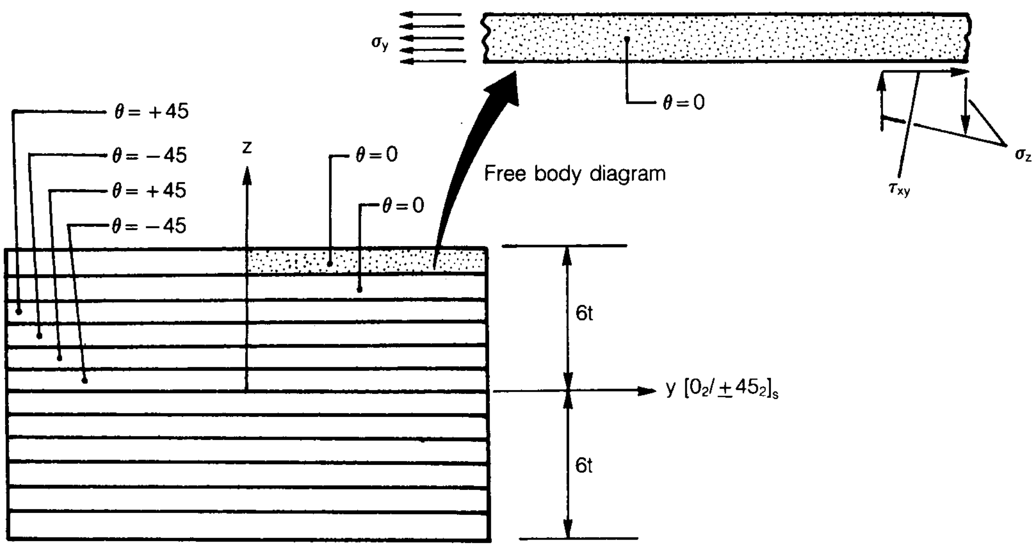


Fig. 7.3.13 Interlaminar Geometry and Stresses

Chapter 8.0

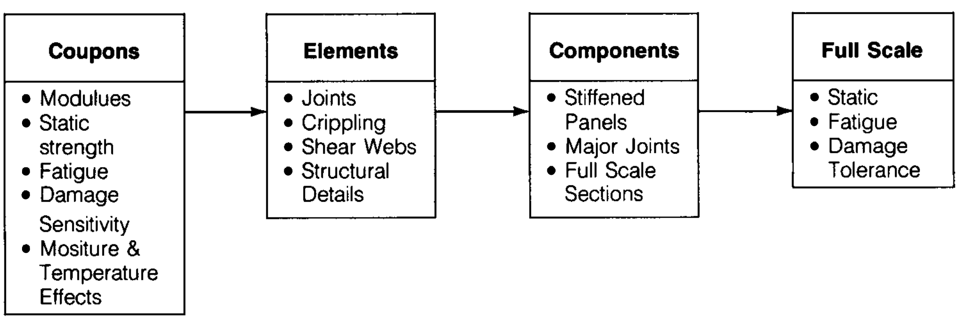
STRUCTURAL TESTING

8.1 INTRODUCTION

Airframe structure design requires a continuing assessment of structure function to determine whether or not the requirements have been satisfied. The expected service performance must be assessed before the structure enters the service environment. This assessment is the structural testing which will ensure and substantiate structural integrity per certification criteria for either civil or military requirements. The basic ''building block'' approach, shown in Fig. 8.1.1, for testing of anisotropic laminate structures should be established in the early stages of development because the validation process for composite structures is very dependent on testing of all levels of the fabrication process.

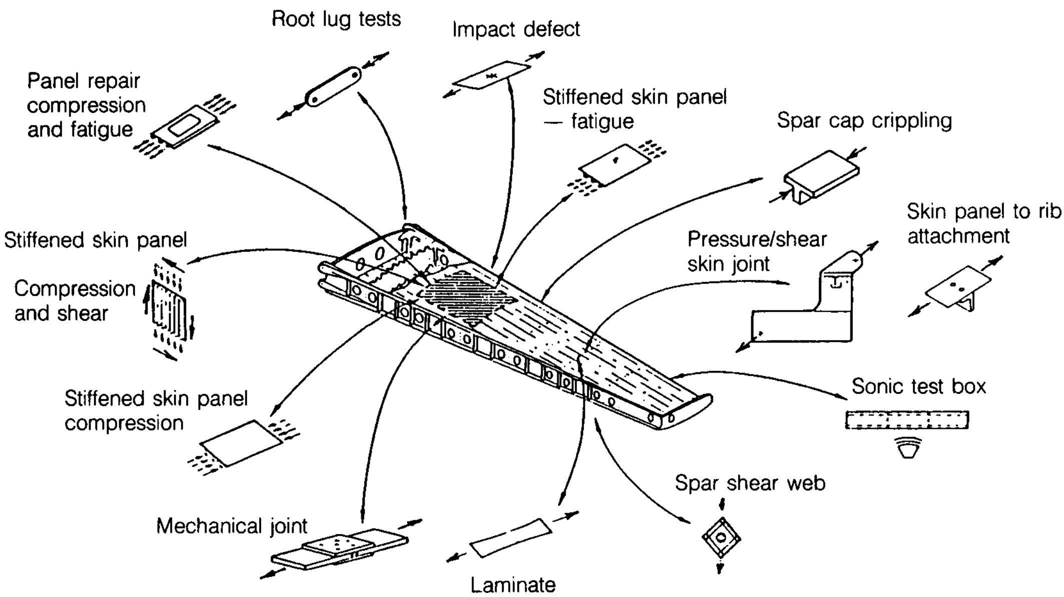
Composite structural testing is similar to most metal structural testing (the majority of metal testing procedures are applicable to composite structures) in that it requires knowledge of design and analysis. The difference is that composites behave anisotropically and need thorough experimental testing, not only of the structure as a whole, but also of test specimens at the coupon, element, and component levels.

Design with composite materials requires a knowledge of lamination theory and appropriate failure criteria (see Chapter 7) as well as related analyses. These analyses must deal with the new set of material properties that result from the making of a laminate. Laminate properties test results are not useful to the engineer until the data is reduced, translated into design allowable, and then reported in a standard format that is easy understand.



(a) Building block test sequence

Fig. 8.1.1 Building Block Testing Approach



(b) Example

Fig. 8.1.1 Building Block Testing Approach (cont'd)

This chapter presents a comprehensive overview of the structural testing of composite materials and hardware to assist in validating design predictions and verifying methods of analysis. It is not possible to foresee and describe all the situations which might arise in a given experimental project. This chapter will not provide detailed description of those tests which are performed on a routine and highly repetitive basis. The engineer will instead be directed to the references which list appropriate published specifications and standard testing technical reports. These references will better serve to alert the engineer to potential difficulties, give deepening understanding of their purpose and limitations, and explain how the data must be reduced.

The purpose of a structural test program is to establish failure modes, demonstrate compliance with criteria, and correlate test results with theoretical predictions and thus assure confidence in the part or overall airframe structure that it will perform satisfactorily throughout its service life. It is therefore important that the structure's constituent components and materials are studied under an encompassing range of service conditions before a program is locked into a production design. For example, expensive redesigns may be avoided by an early screening of matrix materials to assess moisture degradation effects. A broad range of material and component characterization tests should be completed on basic hardware, and then proof and ground tests performed on prototype structures. A large number of tests are required to satisfy these requirements.

It is important that emphasis be placed on accurate material property characterization. Modern computer design techniques used in analysis of composite anisotropic materials are extremely dependent on, and sensitive to, the quality of the material property parameters which are furnished by the testing results.

General Considerations

Test engineers must constantly bear in mind the characteristics of composites and must be aware of the following test problems which are unique to composites and cannot be ignored:

- The variations in composite material strengths can lead to increased scatter in test results
- The very low interlaminar shear and transverse tension strengths of laminates, relative to their inplane strength, lead to difficulties in load introduction.
- Special precautions must be taken to diffuse the applied load into composites.
- The free edge effects causes some cross-plied test laminates to exhibit a tendency to delaminate, a failure mode fundamentally different from virtually all isotropic metal structures
- Composite materials are generally brittle, which tends to influence testing as follows:
  - Response is often linear up to failure, thereby easing the measurement problem
  - Small damage resulting from specimen preparation or pretest handling may result in high scatter, i.e., low test values
  - A small misalignment of fibers relative to the test direction may also cause significant data scatter
  - Lack of ductility around fastener holes means extra care must be taken in creating joints which diffuse the load into the structure.
- Testing to assess moisture-induced degradation is of course unique to composite and adhesive joints
- Non-visible damage that results from impact, in-process procedures, NDI limitations, etc. must be addressed in the material properties

Testing Standards

Most of the effort devoted to materials testing is directed toward the testing of small samples in order to evaluate the properties required for use in lamination theory analysis. While new improved methods are always being evaluated, much of this type of testing follows the standard procedures described in Ref. 8.1 and includes:

(1) The measurement of unidirectional coupons to determine the material moduli, Poisson's ratio, and failure stresses and strains in the direction of the two primary axes (0° and 90°). This refers to the fiber direction and normal to the fibers in tape while for cloth it is the warp and fill directions. This can only be accomplished when the test specimens have all plies laid up in the same direction and considerable effort is devoted to providing a uniform strain in the test section.

(2) Relative to testing to obtain the strength of cross plied laminates, using lamination theory and the failure criteria derived from the above unidirectional materials testing, a good estimation can be made of laminate failure. However, the testing of typical laminates is required to evaluate the accuracy of such predictions. Both material and laminate tests are repeated often enough to provide sufficient data to compute the "B" basis values of strength, stiffness, etc. The combined effects of moisture and temperature must be included in these tests. The degradation of matrix -dominated properties such as compression strength, due to moisture absorption is very important.

It must be remembered that laminate theory is not valid in the immediate neighborhood of joints, cutouts, free edges, simulated damage, and locations where concentrated loads are applied. Thus the material properties derived for use with lamination theory are an insufficient description of material behavior. To account for the behavior of the composite in the areas mentioned, it is necessary to test specimens containing the relevant source of stress concentration. This type of testing must be done on all structural elements tested.

The testing of fiber reinforced composites is so complex that often a new test concept can be evaluated more efficiently with the aid of a computer than by design and fabrication of expensive 'proof-of-method' test hardware which an analysis may indicate would not function suitably.

Types of Structural Testing

(1) Material Qualification Testing

  (a) Its purpose is to determine minimum mechanical properties and manufacturing processing requirements for qualification (see Fig. 8.1.2) to a process specification for:

  - New materials
  - Second source materials (requires approximately 70 coupons for same family materials)
  - Alternate source materials

Requirement	Specification
Basic resin type	Toughened thermosetting
Cure temperature	350 °F
Service temperature	−65°F to 200°F
Uncured resin content	35±2% by weight
Volatile content	0.4% weight
Tack	Within specification
Areal weight	145±5 gm/m^2
Storage life at 0°F	6 months
Out-time at 70% RH and temperature noted	10 days at 80°F max 15 days at 70°F max
Chemical characterization	
High pressure Liquid chromatography (HPLC)	Information only
High resolution Infrared spectrophotometry	Information only
Viscosity profile	Information only

Fig. 8.1.2 Example of Matrix/Prepreg Property Requirements Tests (NASA/Industry Standard)

(b) Mechanical property tests (includes environmental effects):
- Strength requirements
- Stiffness requirements

(c) Manufacturing processing requirements:
- Cure and post-cure cycles
- Raw ingredients
- Resin flow and content
- Tack
- Out and storage time
- Drape
- Volatile content
- Fiber areal weight
- Temperature

It is recommended that a chart or plan be developed which summarizes the material qualification tests to be conducted for each new tape or bidirectional fabric composite material that is tested. These tests include the type of tests defined by standard testing methods and specified in each project. Specimen configurations should be defined. The numbers and types of tests for fabric laminates are the same as for tape laminates except for the number of plies used in the specimens. For fabric laminates each ply consists of fibers oriented at 0°/90° or ±45°. Fabric plies are thicker than tape plies and therefore fewer plies are required for equivalent laminate thickness.

The level of moisture conditioning will be determined on hole notched specimens from each batch (e.g., minimum 3 to 5 specimens per batch) used for the moisture conditioning tests. The fabricated specimens will be soaked in 160°F demineralized water up to 12 weeks. Each week the specimens will be removed and weighed. A plot of the percentage of moisture gain versus exposure time will be made to determine the percentage weight gain corresponding to approximately the 2/3 (67% is the real world case and should be used) saturation level for the wet tests. The same weight gain should be used for the moisture (wet) condition tests discussed in the subsequent sections of this chapter.

There are other recommended method for moisture conditioning which is accomplished in a moisture controlled chamber under 160°F temperature with 85% relative humidity (RH) condition.

(2) Coupon tests (static/fatigue-durability):
- Establish lamina material properties
- Establish laminate design allowable (design criterion varies for particular applications)

(3) Element and component tests (static/fatigue-durability) — Used to verify the structural adequacy of a particular structural configuration (e.g., co-cured panel, sine-wave web, beaded web, etc.)

(4) Full-scale tests (static/fatigue-durability) — Full-scale airframe, empennage stabilizer, canard, control surface, pylon, etc., is tested.

(5) Retrofit Test Program
A retrofit program is one in which the design objective is to substitute a composite part or component(s) for existing metal counterpart(s). In material substitution, the composite laminate thickness and stiffness will closely duplicate those of the metal part being replaced. A large number of tests should be performed to satisfy design requirements, but the expense will preclude such tests on most retrofit programs. Standard coupon testing is usually performed to either verify the allowable with minimum testing or to develop an allowable with substantial testing. The selected retrofit part should be designed and tested as a quick way to start verifying:
- The structural design without incurring costly design risk
- The manufacturing approaches and structural integrity

(6) Other tests:
- Environment (part of Material Qualification Testing)
- Lightning strike (see Section 6.3 in Chapter 6)
- Damage tolerance (part of Material Qualification Testing)
- Hydraulic ram (see Fig. 8.1.3)
- Ballistic impact
- Fluid resistance test (functional test)

Moisture and Temperature Effects

Composites are sensitive to both temperature and moisture absorption from service environment exposure which must be accounted for by means of the analysis and tests required for certification. The induced stress-strain reductions are caused by the structural response at the following levels:

(a) At the ply-level (or coupon level, see Fig. 7.2.1) — The need for testing at this level is due to the physical compliance of the laminate which is usually accounted for by ply-level properties. These moisture and temperature effects are accounted for in the stress analysis of the structure by using a reduced allowable.

(b) At the structural element or component level — The actual allowable strength of the composite element results from the use of different ply orientations, and the inherent structural redundancy of a high performance complex airframe structure. This is verified by testing.

The selection of environmental conditions, the relative humidity and temperature, to be used for the conditioning process is a function of the exposure conditions of the actual hardware and the time available for the test program.

Environmental conditions to be tested are:
- Temperature — cold, room and high
- Moisture content — 67% saturation and dry

Data Documentation

Test data involves identifying and tabulating the appropriate:
- Specifications
- Materials and processing records

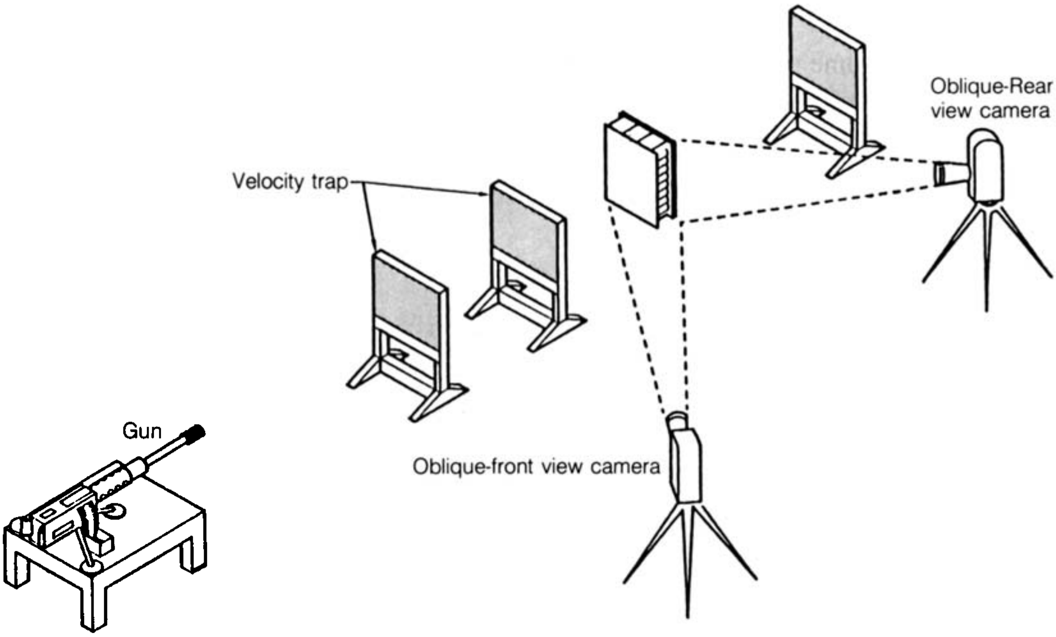


Fig. 8.1.3 Hydraulic Ram Testing Setup

- Inspection records
- Specimen preparation records
- Test data logs
- Stress-strain curves

Complete traceability is necessary for individual specimens regarding their:

- Locations in the laminate
- Inspection and processing records
- Material identification — prepreg batch number and fiber lot number

Obtaining this level of traceability requires proper organization of the test program from the beginning, proper monitoring in progress, and collection of all documentation for data analysis. Controlling and monitoring all these steps virtually assures that most of the test data will be acceptable, thus minimizing retest requirements. Correlation of experimental and analytical data is one of the basic parts of data analysis.

It is easier to record test data and information than to reconstruct it later; the test report should include:

- A complete description of the material, including:
  - Code numbers
  - Lot numbers
  - Run numbers
  - Any physical test data

- A description of the technique used to fabricate the parts especially noting layup room:
  - Out-time
  - Moisture conditions
- The number and orientation of plies
- The specimen fabrication techniques including:
  - Cutting methods
  - Moisture conditions
  - Storage times
- The measured dimensions of each specimen
- The instrumentation and methods of attachment, including storage time
- All information relating to deliberate exposures
- The test environment, including temperature and humidity
- The type of machine, the grip, and the speed of the test
- All the calculated values of modulus, strength, strain-to- failure, and Poisson's ratio, including any test points that were discarded and the reasons for discarding them
- The type of failure

8.2 DURABILITY AND DAMAGE TOLERANCE TESTS

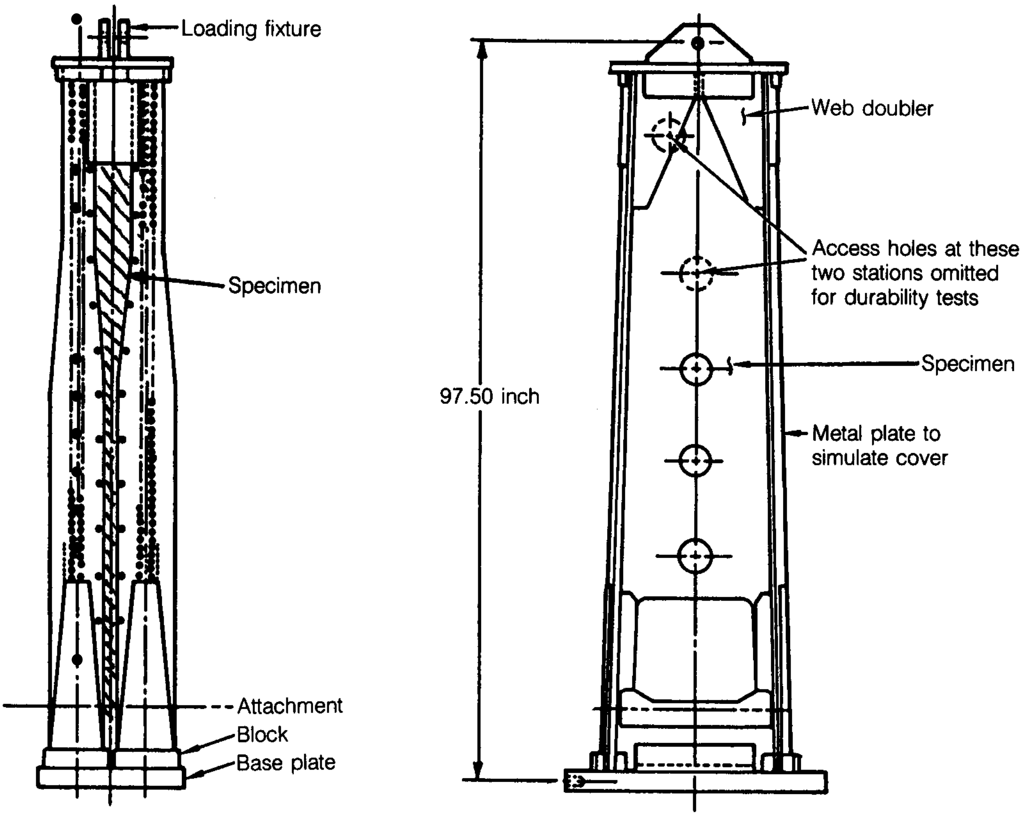
An airframe must satisfy static, durability, and damage tolerance requirements to achieve structural integrity and reliability. Test data reveals that:

- A laminate can lose approximate 60% of its undamaged static strength with impact damage that is essentially non-visible
- The loss of strength in the cyclic-load durability test is also greatest from impact

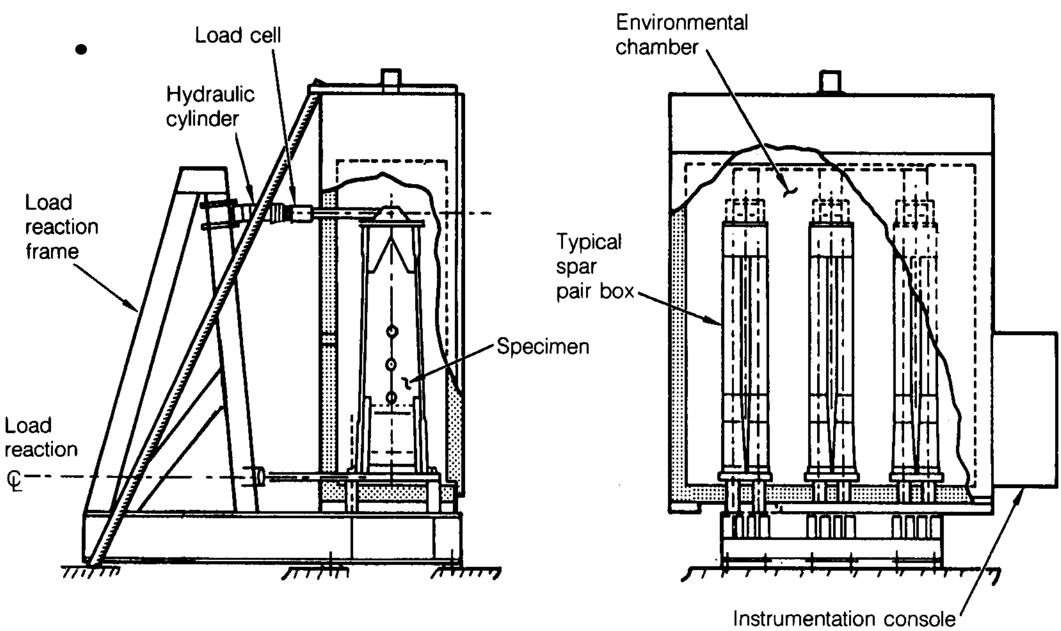
Durability (Fatigue) Testing

Durability testing is generally understood to be fatigue testing, either constant cycle or longtime-cycle spectrum load testing. However, with composite materials, the effects of environmental exposure on static and dynamic behaviors must be considered. Thus, durability testing for composite structures becomes a function of load cycling and environmental exposure. Airframe durability testing, as shown in Fig. 8.2.1, is accomplished in a complicated and sophisticated manner, using a flight-by-flight real-time loading spectrum related to aircraft lifetime and, concurrently, environmental exposure based on flight temperatures and ground-based moisture environments. In addition, accelerated flight spectrum loading and accelerated moisture/temperature environments have been used to simulate real-time testing, with some success and some failure in correlation.

For fighter airframes, accelerated testing to 4 to 5 lifetimes under the worst environmental conditions would simulate one lifetime of real-time testing. Design requirements for fighter airframes generally state that the durability testing must simulate 4 real lifetimes; therefore, accelerated testing would require at least 16 lifetimes.

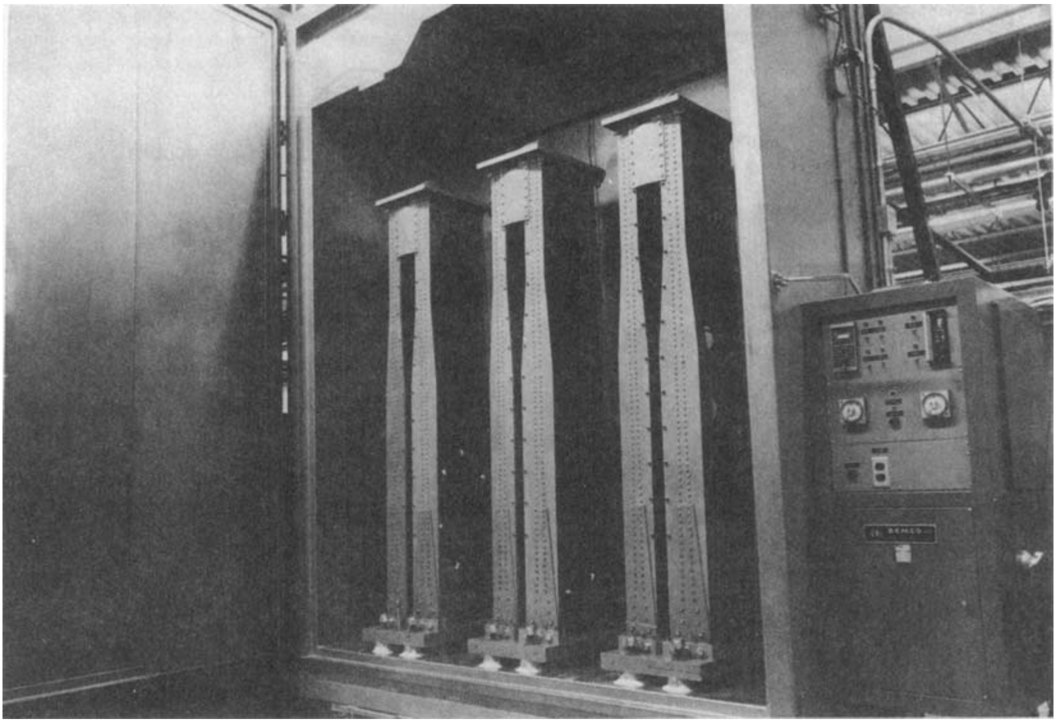


(a) Spar durability test specimen



(b) Spar durability test setup

Fig. 8.2.1 Spar Segment Durability Test For L-1011 Vertical Fin Box (Graphite/Epoxy)



By courtesy of Lockheed Aeronautical System Co.

(c) Specimen in the environmental chamber

Fig. 8.2.1 Spar Segment Durability Test of L-1011 Vertical Fin Box (Graphite/Epoxy) (Cont'd)

Fiber-dominated laminates are considerably more efficient in load-carrying ability than are matrix-dominated laminates; however the latter are sometimes needed, for multidirectional loadings and damage tolerance requirements. It is generally assumed that matrix-dominated laminate design is governed by durability strength, whereas fiber-dominated laminate design is governed by static strength. Therefore, durability testing for structural integrity verification of matrix-dominated laminates must necessarily include bonded joints.

The effects of cyclic loading on ''current'' carbon/epoxy composite structures have generally been shown to be non-critical due to low sensitivity to stress concentration at the low level fatigue load spectrum. The load threshold at which composites become sensitive to cyclic loading is a very high percentage of their static load. This threshold load is very high and most aircraft do not experience cyclic loads that even begin to approach their ultimate loads.

Damage Tolerance Testing

Damage tolerance testing is significantly different for composites than for metals. Damage tolerance in metals is related to the rate of propagation of a crack of a given size and location, whereas damage tolerance in composites is primarily dependent on resistance to impact.

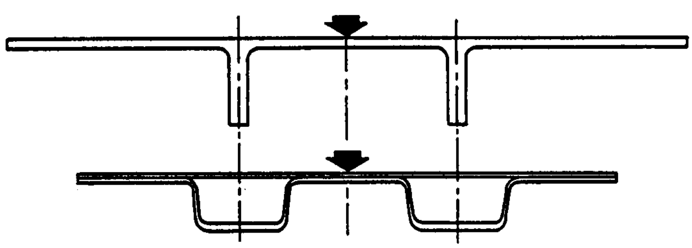
Structures made with composite materials must be designed to support design loads after impact that has a reasonable probability of occurring during fabrication or service life. To define a strain allowable to account for impact damage compression stress is somewhat akin to defining fatigue allowable for metal structure (tensile stress is critical). The fatigue allowable are selected based on limited tests and previous design experience. However, final fatigue substantiation is based on durability (fatigue) tests conducted on full-scale components or the complete airframe. Compression tests are conducted on impact damaged coupons to select preliminary compression design stress allowable, and then compression tests of impact damaged structural panels and subcomponents are conducted to substantiate the design allowable.

To define design allowable for impact damage, tests are conducted on flat laminates loaded in compression. These may have varying amounts of impact damage, dependent on the panel thicknesses and the damage tolerance requirements for damage visibility and maximum impact energy. The panels must be large enough to nullify size effects, e.g., 10 inches (25.4 cm) × 12 inches (30.48 cm). The results are representative of damage to the areas of the structure between reinforcements (e.g., stiffeners). The effect of impact damage where reinforcements are attached to the skin or the effect on the reinforcements themselves is evaluated based on tests of reinforced panels.

Since strength and damage sustained may be a function of layup configuration, several variations of each laminate should be tested. The effect of environment will also need to be evaluated with tests for given conditions. Some tests should be conducted with higher impact energies to determine the trend of data for wider damage widths. It will also be necessary to conduct sufficient cyclic tests to ensure that no detrimental damage growth will occur during the expected service life.

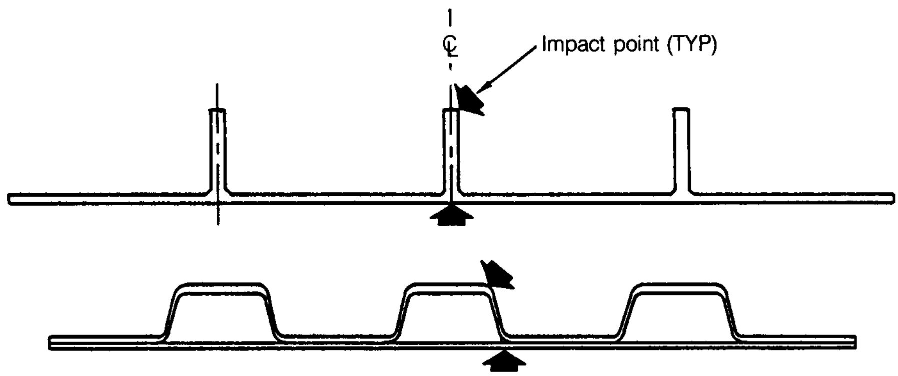
Impact damage testing of a laminate is done on laminate panels, as shown in Fig. 8.2.2, using a support fixture with either simple or fixed edge support. Fig. 8.2.3 shows an example of integrally-stiffened panel in a test fixture. The damage requirements vary considerably, depending on mission and lifetime requirements. The requirements for a typical military composite structure are as follows (Ref. 7.34 and also see Section of 7.4 in Chapter 7):

(a) Low level impact damage:
- An impact of 6 ft-lbs (8.4 J) from an impactor with a 0.5 inch (12.7 mm) diameter hemispherical head
- The damaged laminate should have the capability of carrying static ultimate loads.



(a) Impact on skin

Fig. 8.2.2 Impact Locations On Test Panels Prior To Post-Impact Compression Test



(b) Impact on stiffener and on skin-opposite-stiffener

Fig. 8.2.2 Impact Locations On Test Panels Prior To Post-Impact Compression Test (cont'd)

(b) High level impact damage tests:
- An impact of 100 ft-lbs (140 J) from an impactor with a 1.0 inch (25.4 mm) diameter hemispherical head; or an impactor which would not cause a dent deeper than 0.1 inch (2.54 mm)
- The damaged laminate should have the capability of carrying static limit load

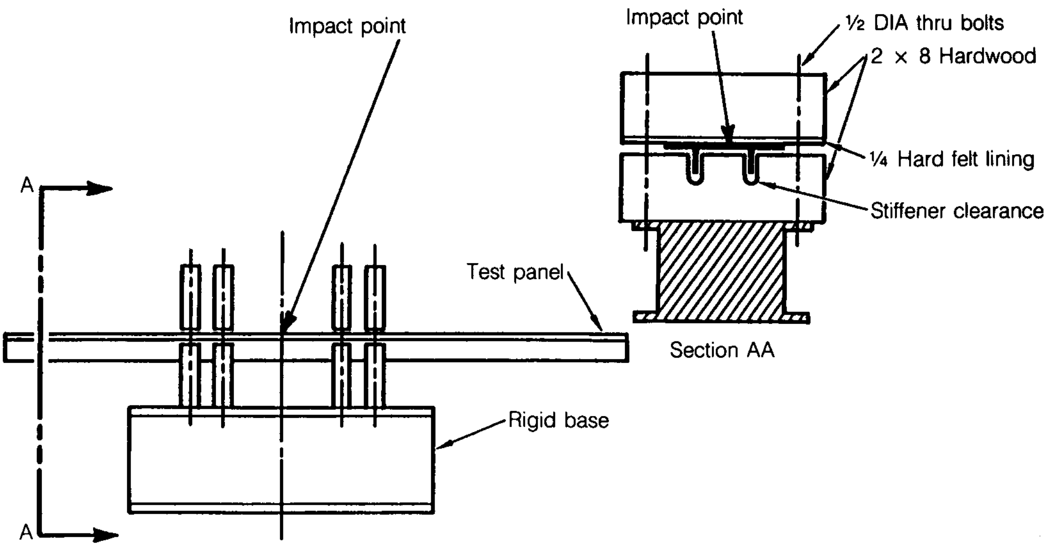
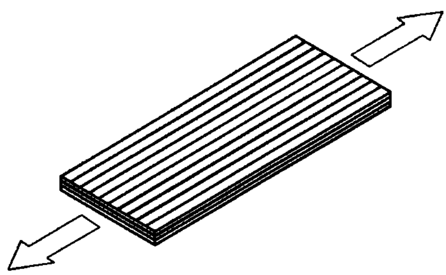


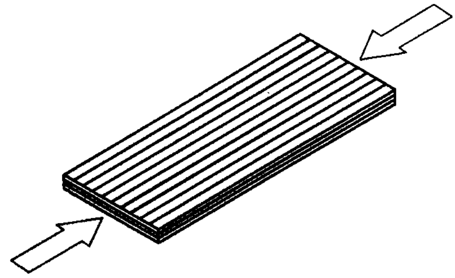
Fig. 8.2.3 Arrangement For Impact Tests Of An Integrally-Stiffened Panel

8.3 COUPON TESTS

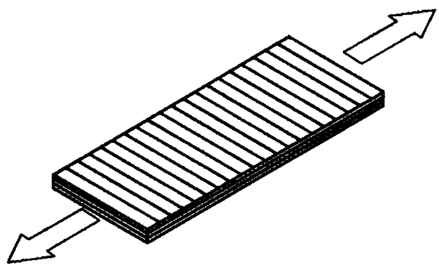
Single ply (lamina; tape or fabric) properties are obtained experimentally from multi-ply, unidirectional laminate specimens where all plies have the same orientation. For tape laminates with all fibers aligned in the same direction (also may consider to test cross-plied laminates to obtain unidirectional properties), the ply properties needed for design (see Fig. 8.3.1) are:



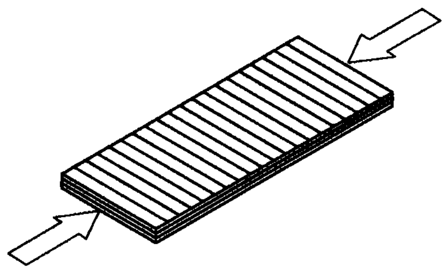
(a) 0° Tension



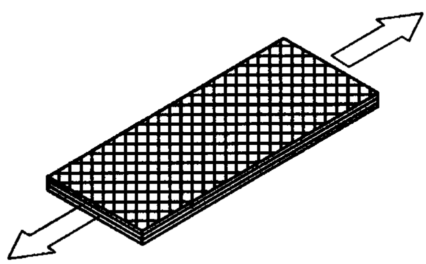
(b) 0° Compression



(c) 90° Tension



(d) 90° Compression



(e) ±45° Tension

Fig. 8.3.1 Ply Level Tests

- Ultimate strength values
- Elastic constants
- Poisson's ratio values

Test coupons that are designated to be weighed during the conditioning process should be weighed immediately after fabrication. All of the coupons are then stored in a dry, desiccated chamber prior to conditioning. It is important that the fiber volume (see Fig. 8.3.2) and void content of each coupon be known. Moisture is absorbed by the matrix, so the percentage of matrix in a given coupon will affect the amount of moisture absorbed. The size and concentration of voids present in the coupon must also be known. The relative humidity in the controlled chamber will determine the maximum moisture content of the conditioned test specimens (see curve shown in Fig. 6.2.1).

Property	Effect of fiber volume percentage (FVP) on the mechanical properties of the following laminates.*			
	$[0]_{nt}$	$[90]_{nt}$	$[\pm 45]_{ns}$	$[(\pm 45)_5/0_{16}/90_4]_c$
Ultimate Strength	Varies directly with FVP	Not sensitive to FVP	Varies directly with FVP	varies directly with FVP
Ultimate Strain	Not senstive to FVP	Not sensitive to FVP	Not sensitive to FVP	Not sensitive to FVP
Prop. Limit Stress	Varies directly with FVP	Not senstive to FVP	Varies directly with FVP	Varies directly with FVP
Prop. Limit Strain	Varies directly with FVP	Not senstive to FVP	Varies directly with FVP	Varies directly with FVP
Poisson's Ratio	Not sensitive to FVP	Not sensitive to FVP	Varies Inversely with FVP	Not sensitive to FVP
Modulus of Elasticity	Varies directly with FVP	Varies directly with FVP	Varies directly with FVP	Varies directly with FVP

*The above deductions are valid for both tensile and compressive properties.

Fig. 8.3.2 Effect of Fiber Volume Percentage (FVP) on the Mechanical Properties of Test Laminates

Load introduction tabs are used to hold the test specimen in place and to insure that local stresses remain low so that the test specimen fails where it is designed to fail. Specimen tab design should take the following into considerations:

(a) Tabs for graphite/epoxy specimens are made of woven glass/epoxy or suitable graphite/epoxy prepregs that have been premolded under suitable pressures and temperatures into solid laminate sheets.

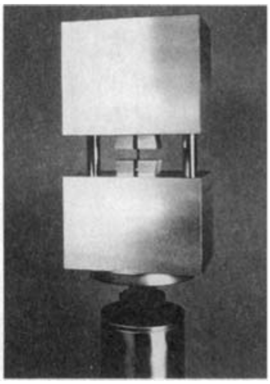
(b) Unidirectional glass/epoxy and graphite/epoxy tape laminates have been successfully used as tab materials, but they are usually more expensive.

(c) Metal tab materials such as titanium or steel may be used where higher tab-to-specimen bonded strengths are required.

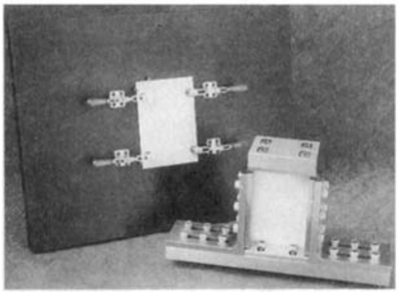
(d) Both aluminum and magnesium tab materials are acceptable on small elements such as coupon or small articles where a small mismatch of CTEs between the tab and test specimen is acceptable.

A table should be made which summarizes the tests to be conducted to determine the lamina properties for tape and bidirectional fabric laminates. The specimens can be cut from the same laminate plate, but oriented so the loading direction is in the 0° direction and 90° direction as required.

Fig. 8.3.3 shows test fixtures for composites. The strain measurement can be obtained by using either strain gages and/or extensometers (see Fig. 8.3.4). Testing at elevated temperature requires certain modifications from standard practice because of the increased drying rate caused by heating. Ideally, testing in an environmental chamber (see Fig. 8.3.5) would eliminate the requirement to make undesired compromises by maintaining the moisture in the saturated specimen.

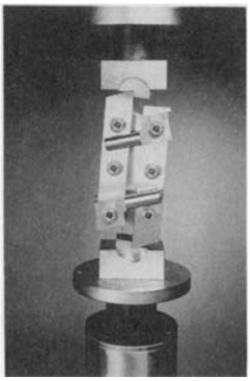


(a) IITRI compression fixture has a unique design that reduces friction for more accurate test results.

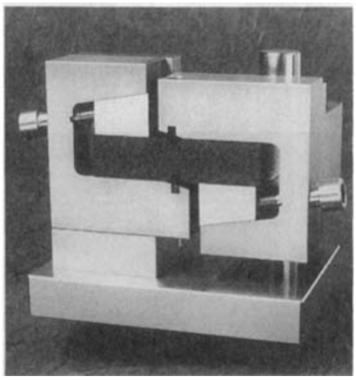


(b) Boeing compression-after-impact fixture allows a reduction in material allocation.

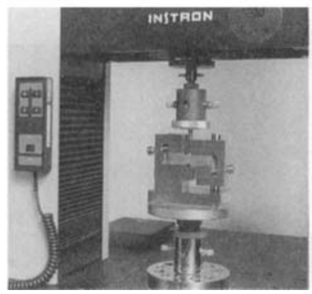
(c) Split disc tensile fixture is used for testing reinforced plastic rings.



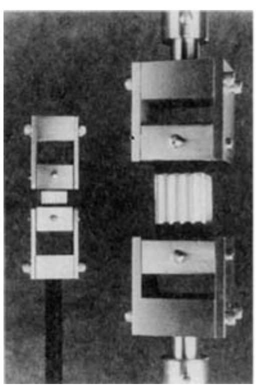
(d) Two-rail shear fixture is recognized industry-wide for consistent shear property determination.



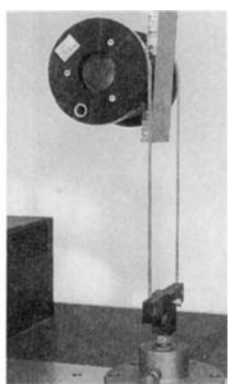
(e) Iosipescu shear fixture allows strength and modulus to be obtained in a uniform shear stress state.



(f) Short beam shear fixture tests shear strength of composite materials. Meets the requirements of ASTM D2344.



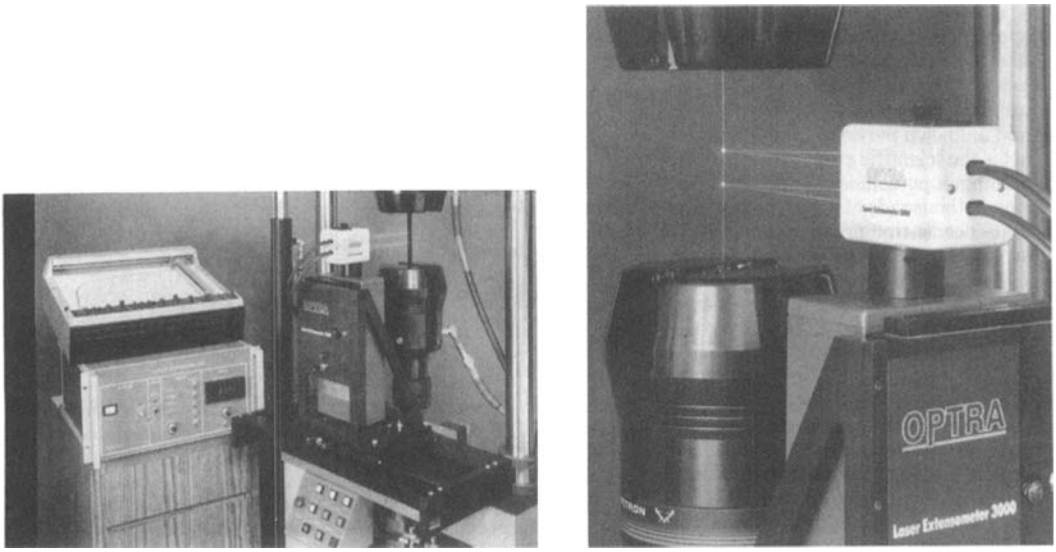
(g) Flat wise tension fixture is for transverse tension testing of composite sections.



(h) Climbing drum peel test fixture measures peel resistance of adhesive bonds.

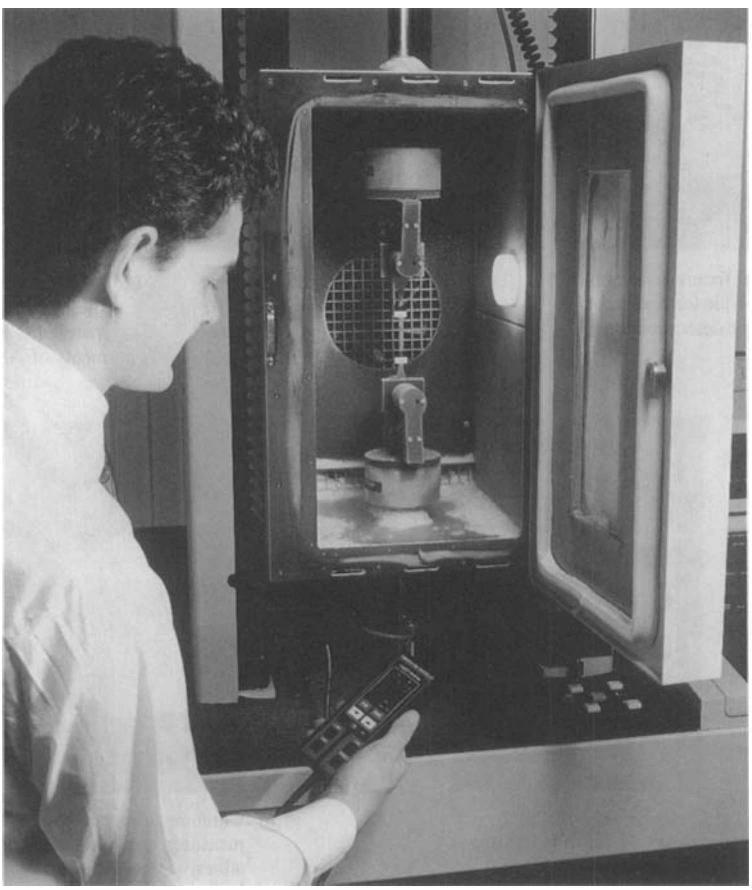
By courtesy of Instron Corp.

Fig. 8.3.3 Various Testing Fixtures For Composites



By courtesy of Optra Inc.

Fig. 8.3.4 Laser Extensometer



By courtesy of Instron Corp.

Fig. 8.3.5 Environmental Chamber System For Coupon Tests [Temperatures Range From $-100°F$ ($-73°C$) to $400°F$ ($200°C$)]

Test results from composite materials tend to exhibit relatively large scatter and thus statistical confidence values are particularly influenced by sample population size; test matrices should be planned to allow for a statistically appropriate number of tests. An approximately 4000 coupons are needed to generate a final set of allowable for both ''A'' and ''B'' values.

There are several basic coupon tests which are routinely performed on composite materials. Two of these tests, tensile and compression, require the specimen to have special load introduction tabs (use on unidirectional specimens for modulus tests only; multi-directional specimens do not use tabs), furnish design data. The short beam shear and flexural tests, provide qualitative data which is more appropriate for acceptance evaluation and assessing processing adequacy. Two other coupon tests are the inplane shear and interlaminar shear tests.

(1) Tensile Test:
The strain measurement of tensile composite specimens can be sensitive to the specimen configuration (configuration does not affect such testing on metals). For unidirectional tape specimens, the recommended configuration is shown in Fig. 8.3.6 which is described in detail in Ref. 8.7

(2) Compression Test:
The measured strain for compression specimens is sensitive to specimen configuration and the fixture used for loading. The specimen must first be constrained from buckling as shown in Fig. 8.3.7; the recommended test fixtures, shown in Fig. 8.3.8 and Fig. 8.3.9, were developed for unidirectional composite test specimens (Ref. 8.8).

(3) Shear Test:
There are numerous shear test methods; some ASTM methods are listed below:

ASTM	Type	Description
D2344	Interlaminar	Short beam shear (3-point) (Ref. 8.9)
D3846	Interlaminar	Short beam shear (4-point) (Ref. 8.10)
D3518	Inplane	$\pm 45°$ tensile test (Ref. 8.11)
D4255	Inplane	Rail shear (Ref. 8.12)

The rail shear method shown in Fig. 8.3.10 tests inplane shear:

(a) Test area of laminate is about 0.5 inch (1.27 cm) wide by 6.0 inches (15.24 cm) long; width may be reduced to avoid buckling

(b) Rails may be bonded to specimen or faced with abrasive paper to avoid failure through bolt holes

(c) Knife-edged bars may replace roller spacers to avoid induced transverse tension under larger shear deflections

Fig. 8.3.11 shows a method of testing interlaminar shear:

(a) Two parallel cuts on opposite faces must both sewer the center lamina at the thickness midplane

(b) Self-aligning grips are used to permit the load to pass through the centerline of the specimen

Tensile test specimen. (1) Fiberglass tabs shall be positioned, both sides, two places; (2) Tabs shall be bonded with adhesives to graphite, dependent upon test temperature; (3) Specimen thickness shall not vary more than 0.076 mm. (0.003 in.) from nominal; (4) Specimen edges shall be parallel to 0.076 mm (0.003 in.); (5) Specimen shall be used for unidirectional (0°) and bidirectional (0°/90°), graphite tape laminates.

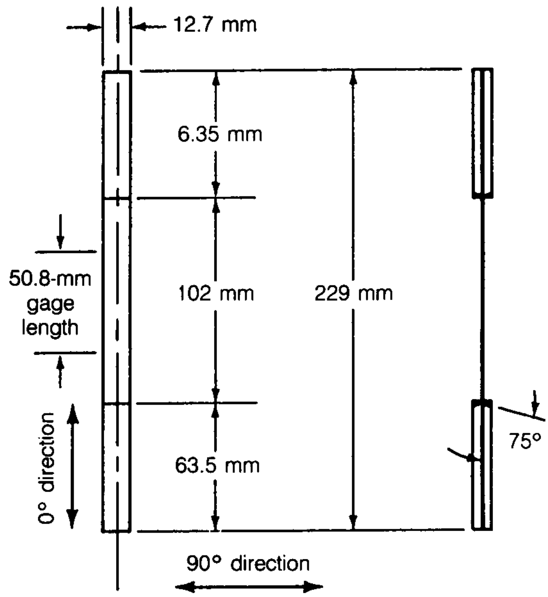
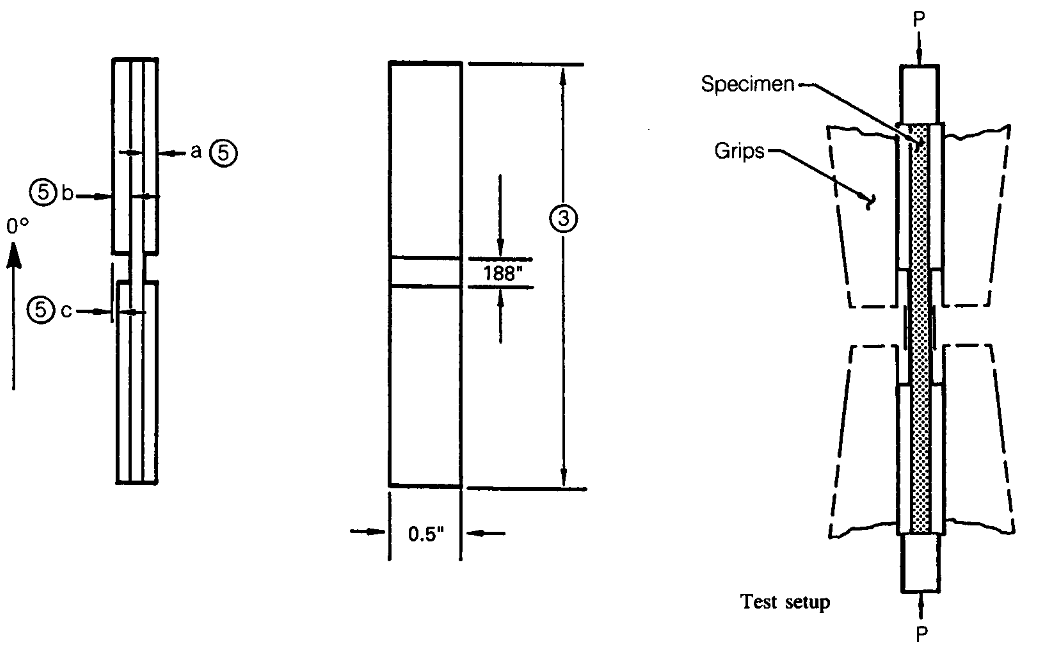
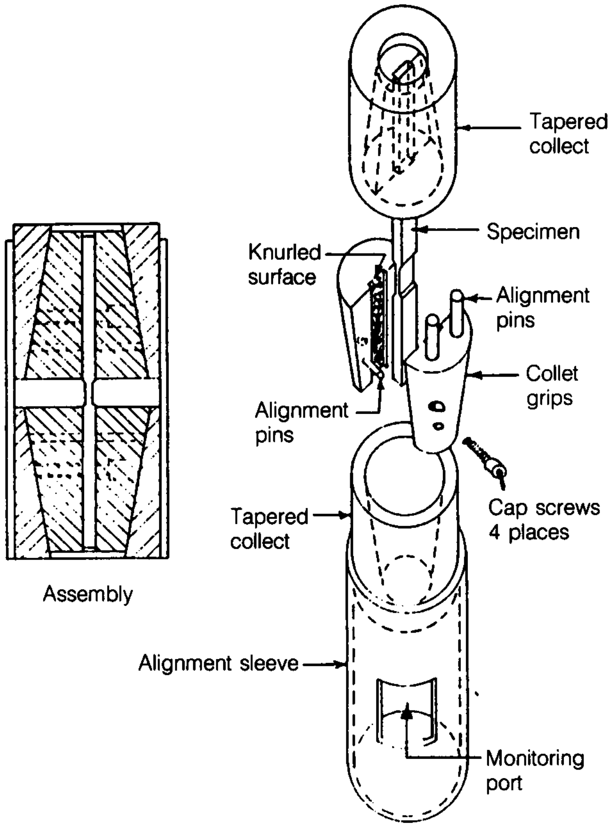


Fig. 8.3.6 Example Of Tension Test Specimen (GR/EP) (Ref. 8.7)

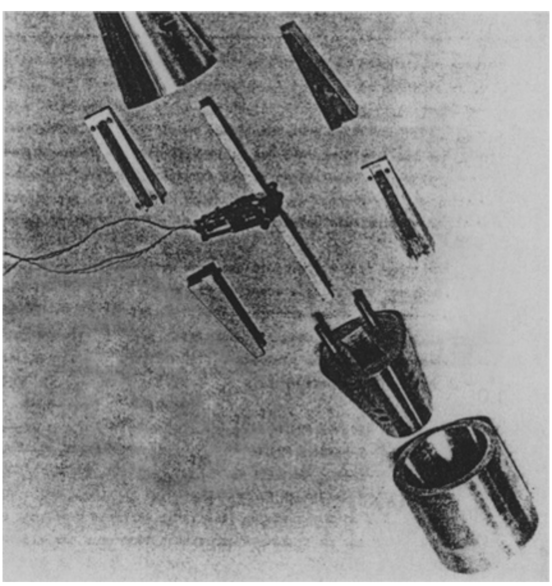


① Test panel layup same as for tensile test specimen
② Support tabs shall be fabricated from a unidirectional graphite material with 0 degree direction in the longitudinal direction of the specimen. Tabs shall be made of 12 plies of grade 190, or equivalent thickness when using other grades.
③ Specimen length is 3.150 ±0.005 inch.
④ Prepare the specimen prior to bonding the tabs by hand sanding the bonding area of the test panel with 150 grit sandpaper and sandblasting or sanding the tab to remove all surface gloss and cleaning thoroughly with acetone or MEK
⑤ A must equal b to within 0.010 inch and c must be less than 0.0025 inch. Rework of the sides of the tabbed specimen may be necessary.

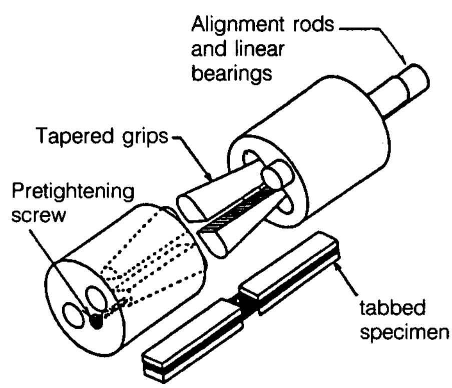
Fig. 8.3.7 Example Of Compression Test Specimen (GR/EP)



(a) Schematic of fixture

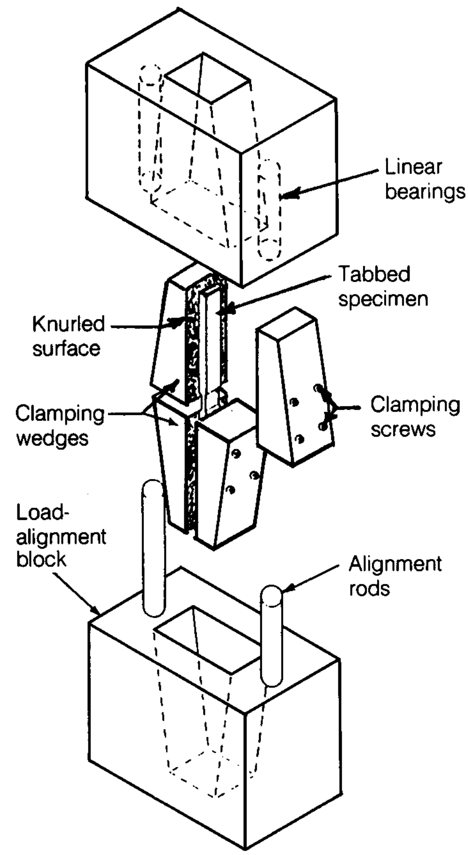


(b) Fixture parts

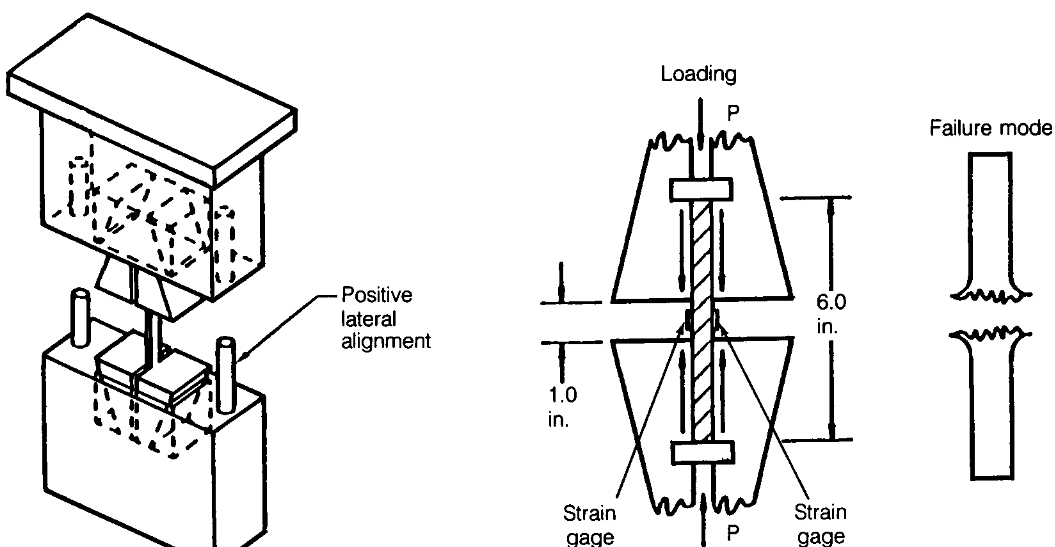


(c) Wyoming modified celanese compression test fixture

Fig. 8.3.8 Celanese Compression Test Fixture

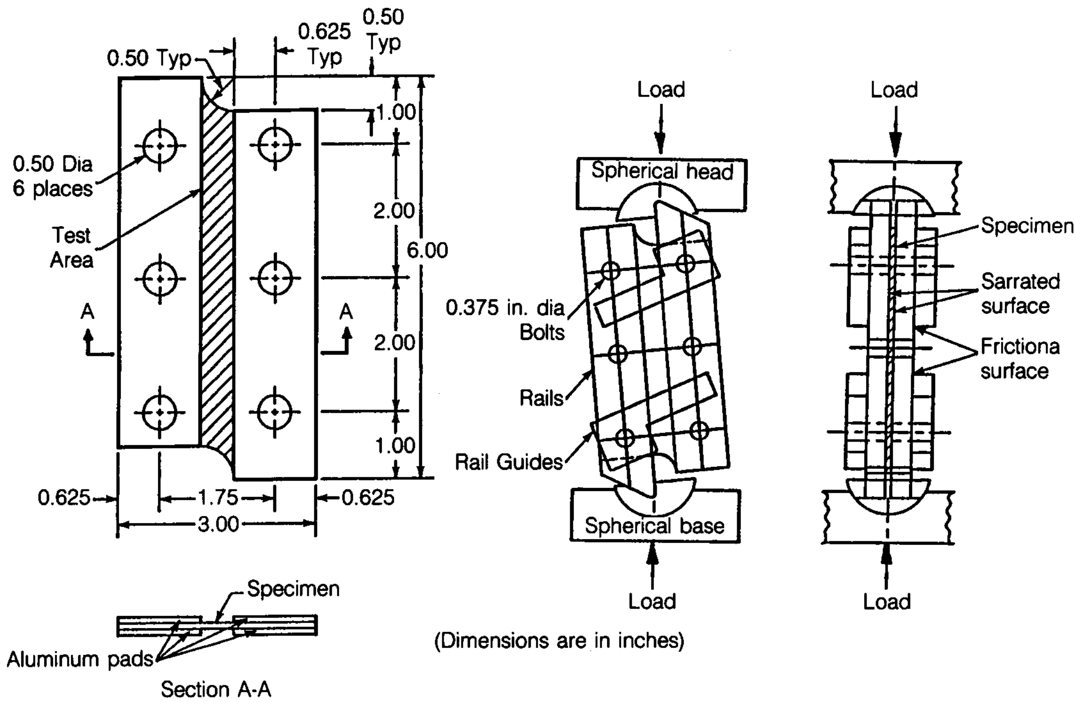


(a) Test fixture

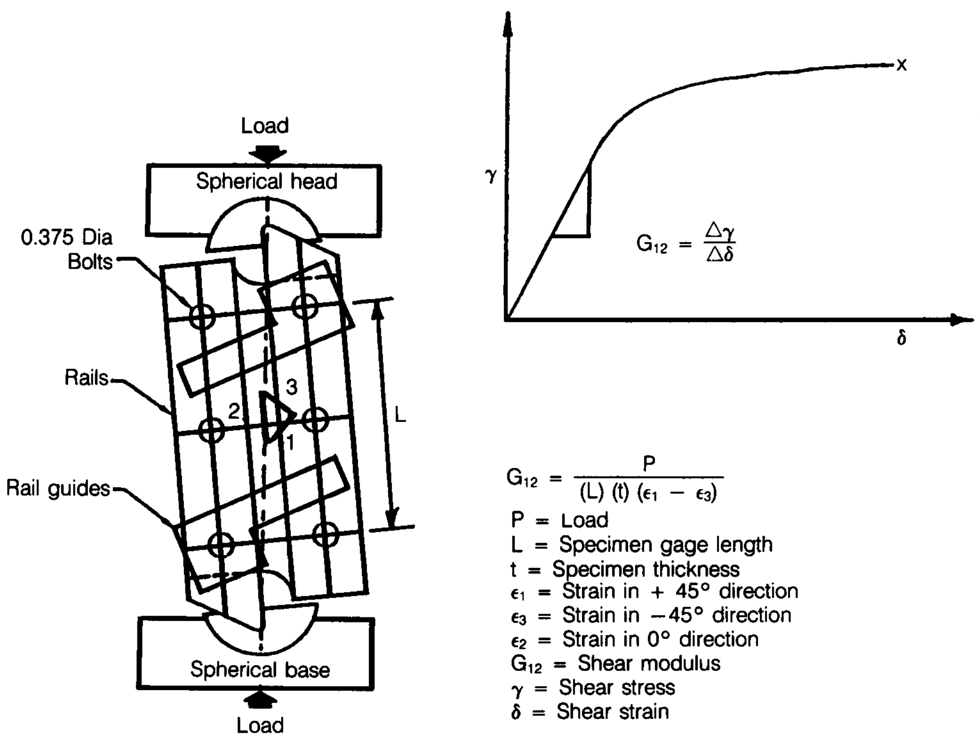


(b) Test setup

Fig. 8.3.9 IITRI Compression Test Fixture

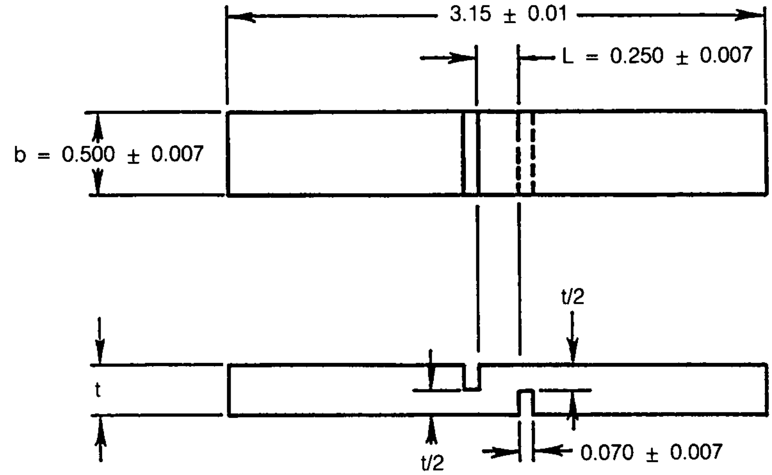


(a) Rail shear test setup



(b) Experimental determination of lamina stiffness shear modulus, G_{12}

Fig. 8.3.10 Rail Shear Test

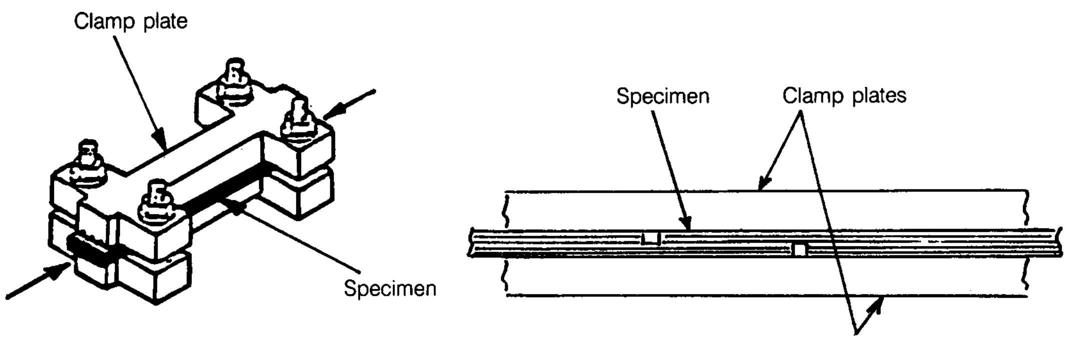


Fabrication
1. Laminate orientation $[45/0/-45/90]_{6s}$
2. Specimen edge parallel and end perpendicular requirement ± 1°
3. Edge finish shall be 32√ in accordance with MIL-STD-IOA
4. Cut specimen notches with an abrasive wheel such that:

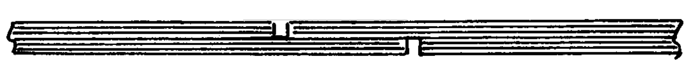
Notch depth = $t/2 \; {}^{+\,0.010}_{-\,0.000}$ and notch penetrates centerply of laminate

Notch corner radius = $0.005 \; {}^{+\,0.001}_{-\,0.000}$

(a) Example specimen (all dimensions are in inches)



(b) Method "A" test (with clamp plates)



(c) Method "B" test (without clamp plates)

Fig. 8.3.11 Interlaminar Shear Tests Using Compression Test Method

(c) Method (A) employs metal plates held snugly against the faces of the test specimen by clamps
(d) Method (B) is loaded to failure without plates
(e) Distance between notches, the laminate thickness and longitudinal modulus affect the test result
(f) ASTM D2345-65T standard:
- Sawcuts to be parallel within 0.030 inch (0.76 mm)
- Depth of sawcut to be equal to:
  - Half the laminate thickness plus one ply, or
  - Half the laminate thickness plus 0.005 inch if number of plies or ply thickness is unknown

The 3-point (Ref. 8.9) or 4-point (Ref. 8.10) short beam test method is a standard test because it is a good and simple measurement and so much data is based on it. The Iosipescu test method [see Fig. 8.3.3(e) and Ref. 8.19] is the most commonly used for testing shear specimens; this method measures both modulus and strength.

Generally, the extensometer is not a reliable instrument to measure inplane shear strain, so strain gages are used.

(4) Flexural Test:

Flexural strength is not considered an intrinsic property, but the test is inexpensive to run and is considered a good quality control test. Ref. 8.13 is the source which is most quoted for flexural test methods; the test arrangement is shown in Fig. 8.3.12.

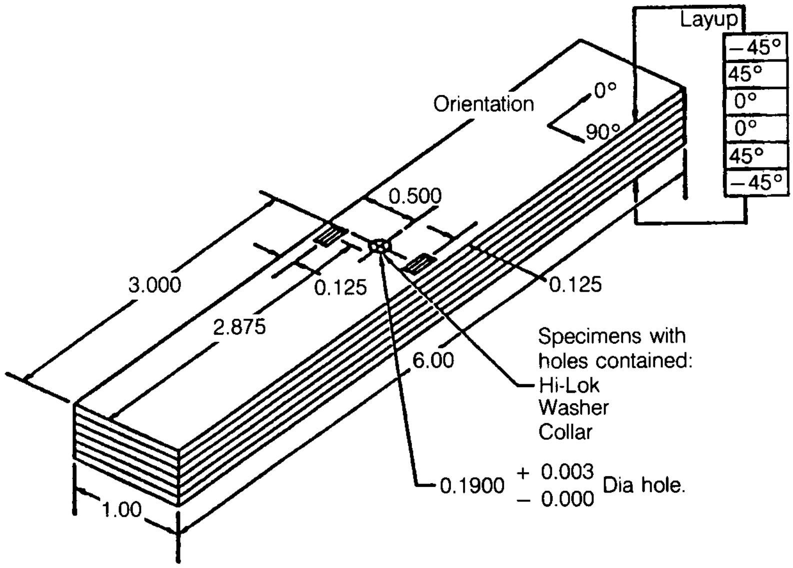
(5) Short Beam Tests:

This test, as shown in Fig. 8.3.13, is useful for evaluating the interlaminar shear behavior of the laminate matrix. Processing variables which will affect the test results are:
- Elevated temperature
- Matrix moisture content
- General matrix condition

This test is generally accepted as a method for obtaining a qualitative measure of the laminate matrix condition rather than as a procedure for generating valid design data.

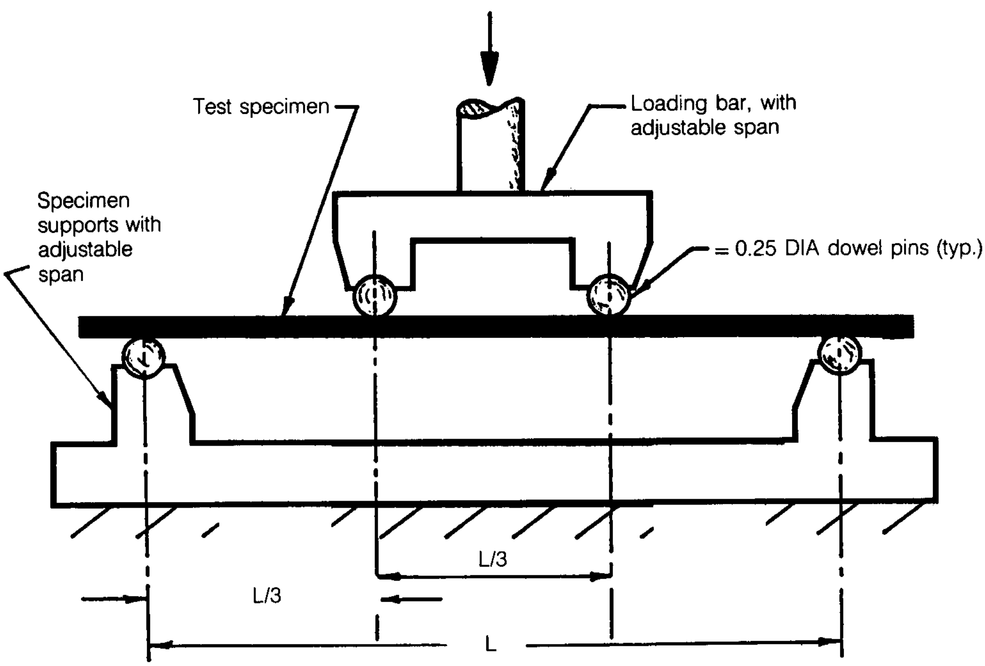
Notched Effect Tests

The notched tension and compression tests are conducted to determine the most damaging combinations of temperature and moisture The effect of hole diameter size (fasteners) on residual tensile and compression strength are evaluated based on the given tests requirements. Tests are conducted for several hole sizes larger than the baseline hole e.g. 0.25 inch (6.35 mm) is the most common baseline with a w/d of 6. The tests are conducted for the most critical environmental condition determined from the tension and compression tests (e.g., see specimen example shown in Fig. 8.3.14). The ratio of hole diameter to specimen width will be the same for all specimens. A reduction factor as a function of increasing hole diameter is determined from these tests.



(Dimensions are in inches)

(a) Test specimen (example)



(b) Test fixture

Fig. 8.3.12 The 4-Point Flexural Test

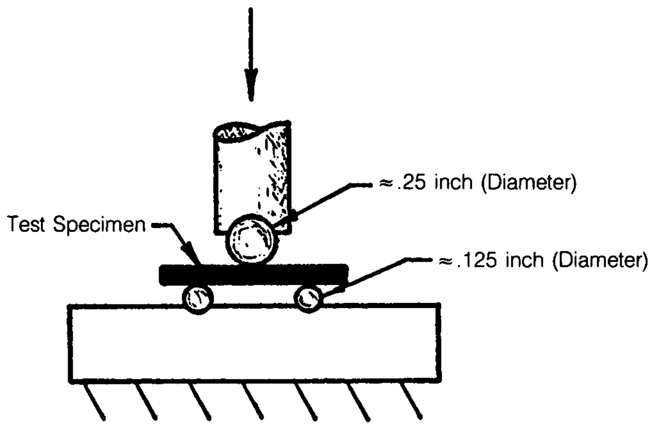
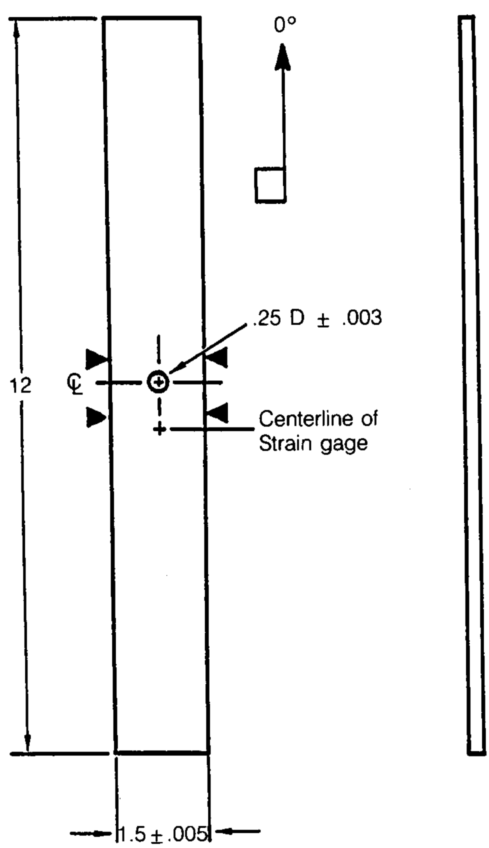


Fig. 8.3.13 The Short Beam Shear Test



1. ▲ AND ▲ Indicate extensometer placement centered relative to the hole. 1-inch extensometer only.
2. Unless otherwise indicated, dimensional tolerances are ± 0.010 inch.
3. Edge roughness in accordance with MIL-STD-10A.
4. Dimensions are in inches.

Fig. 8.3.14 Example Of Open Hole Tension And Compression Specimen

Impact Damage Tests

Fig. 8.3.15 shows a NASA impact test specimen which has a width of 7.0 inches (17.78 cm) and a length of not less than 10.0 inches (25.4 cm) nor greater than 12.0 inches (30.48 cm) . After impact, the specimens are trimmed to a width of 5 ± 0.03 inches (7.54 ± 0.0762 cm) for compression test to failure. However, impact testing on coupons does not give an accurate indication of suitable design properties and is merely used to compare material characteristics. The impact testing to determine design values should be done on components and/or the full-scale test structure.

Fig. 8.3.16 shows an impact machine with a pneumatic clamping fixture and environmental chamber; all data and analysis are controlled by a computer system.

Fastener Bearing and Pull-through Tests

(1) Fastener Bearing tests:

Fastener bearing strength for tape composites is a function of the layup configuration:

- The 100% 0° ply laminate would fail by shear tear out, and strength would be essentially a function of the shear strength of the matrix and the cross-sectional area to the edge of the specimen
- The 100% 90° ply laminate would fail by net section tension and strength would be a function of the matrix tensile strength and the net cross-sectional area

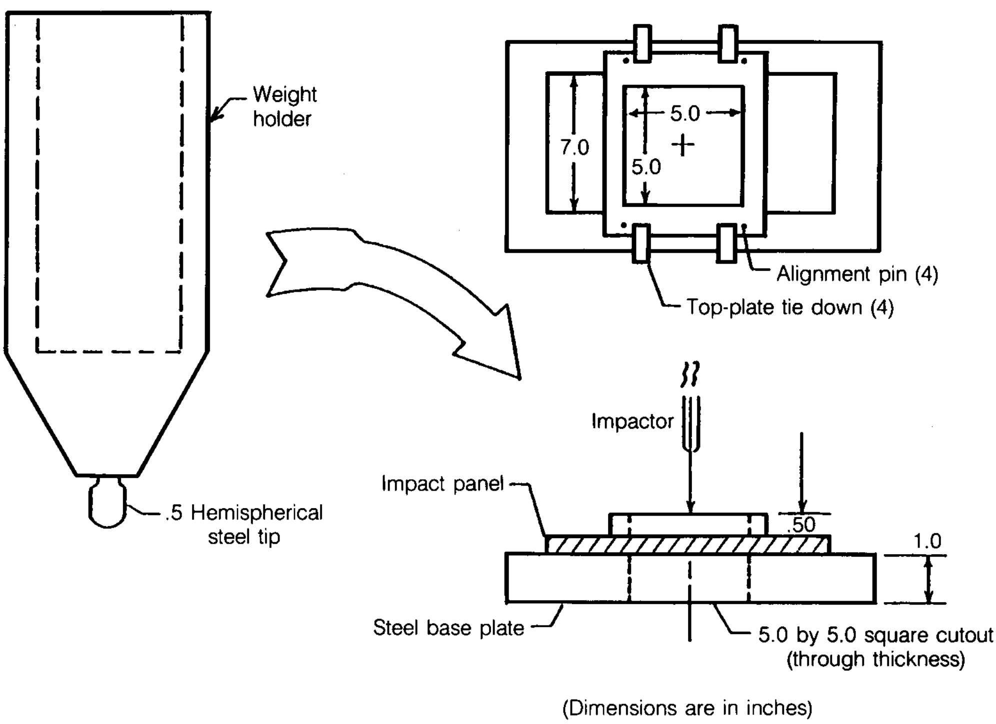


Fig. 8.3.15 NASA Impact Test Apparatus (Ref. 8.14)

By courtesy of dynatup General Research Corp.

8.3.16 Impact Test Machine (Model 8250) With Pneumatic Clamping Fixture And Environmental Chamber

- Both of these are comparatively weak failure modes, multi-directional reinforcement is required if appreciable bearing strength is to be obtained

The bearing strength allowable can be determined from the following equation:

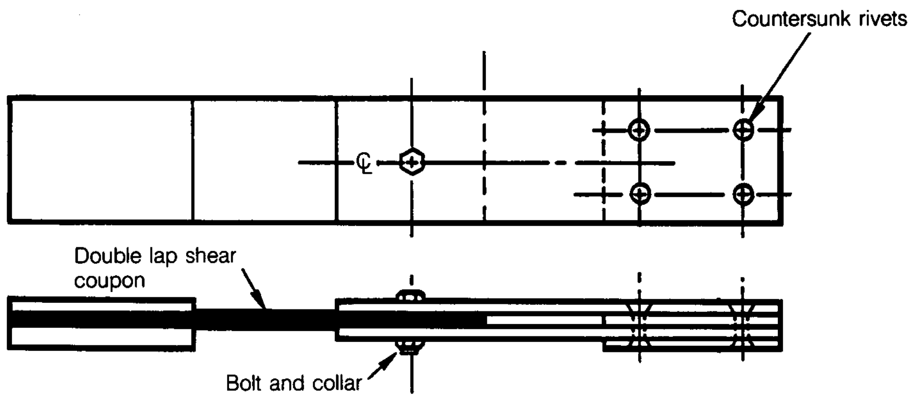
$$F_{bru} = F_{br} \times K_e \times K_{csk} \quad (8.3.1)$$

where: F_{br} — "B" allowable for room temperature dry (RTD) bearing strength of non-countersunk holes

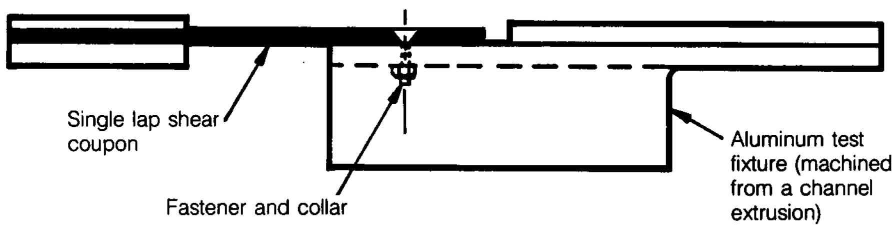
K_e — Environmental correction factor

K_{csk} — Countersunk correction factor

The "B" bearing allowable for a non-countersunk hole and RTD condition is modified by K_e and K_{csk} to account for environment, t/d. w/d, single lap shear, etc. and countersunk thickness for flush fasteners. To determine the F_{br} allowable, tests are conducted on double lap shear specimens, as shown in Fig. 8.3.17(a), made from several different laminate thicknesses. These tests results are pooled to establish "B" bearing allowable. The bearing allowable for a countersunk hole can be tested by using the test fixture shown in Fig. 8.3.17(b).



(a) Double lap shear bolt bearing specimen



(b) Single lap shear countersunk screw bearing specimen

Fig. 8.3.17 Examples Of Fastener Bearing Tests

The test for extreme environmental conditions are used to provide the K_e environmental correction factors. Each specimens used to test environmental conditions should be cut from the same location in the laminates. Then the ratio of the environmental bearing strength to the RTD bearing strength can be used for statistical analysis.

The bearing tests for fabric laminates are similar to the tests for tape laminates.

(2) Pull-Through Tests:

The pull-through test, as shown in Fig. 8.3.18, is conducted to determine the load required to pull fasteners through composite laminates. This property is important for structures subjected to internal pressure loads or to permit buckling of skins (or webs) without failure at fastener attachments. The majority of failures of secondary supports (e.g., secondary tension due to diagonal tension shear buckling effect) are in the form of fasteners pulling through laminates, particularly countersunk fastener heads. Composites are generally weak in pull-through strength.

Pull-through strength is a function of:

- Laminate thickness
- Fastener diameter
- Configuration of fastener head type
- Laminate deflection

Therefore, tests are conducted for several thicknesses , fastener diameters, and fastener head types. Laminate support should supply rigidity to that expected for the structure.

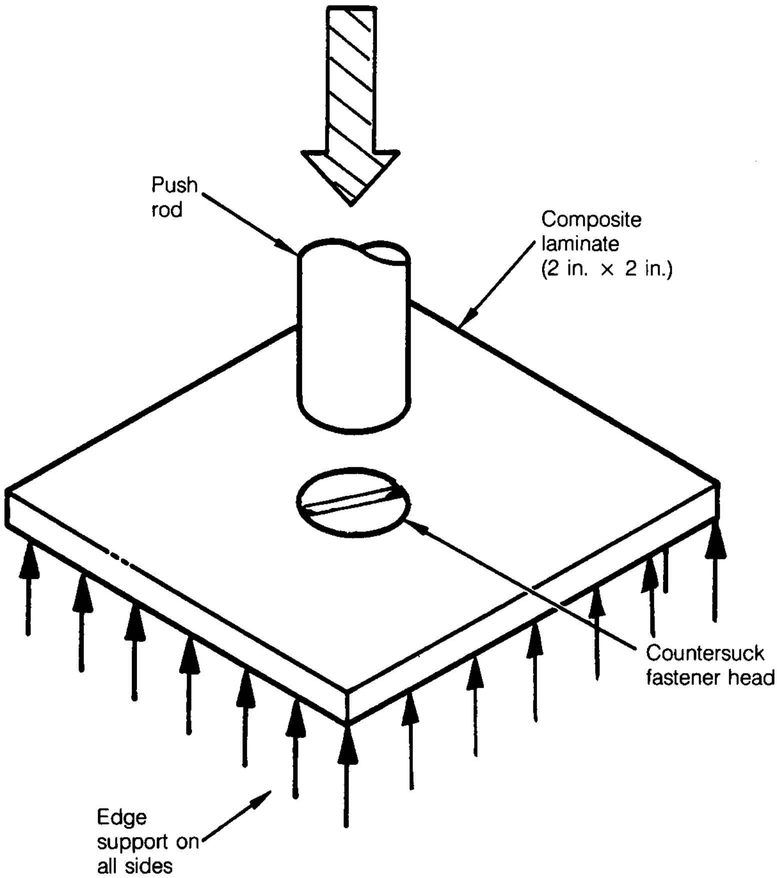


Fig. 8.3.18 Use Push-Through Test Method Of Countersunk Fastener

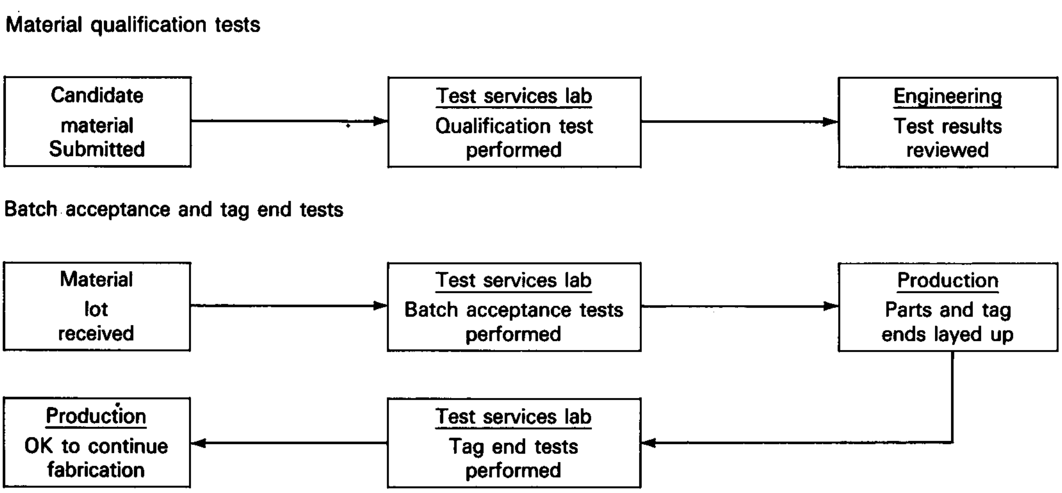


Fig. 8.3.19 Material Qualification Tests Vs. Batch Acceptance And Tag-End Tests

Testing of Aramid Materials

The testing of Aramid materials requires special attention and care for the following reasons:

- Specimens are difficult to machine (fibers rip loose at the edges) because of toughness and a special machine cutter is required
- Longitudinal end tabs often debond before tension failure occurs in the specimen (Aramid has low surface bond strength)
- Compression tests are critical because Aramid has low compression values; for good test results use thicker specimens and make sure there is good alignment of the fibers
- Good test results on inplane shear are difficult to obtain and failure mode is difficult to define due to high fiber toughness

Process Control Testing

Process control testing provides an additional step beyond qualification and acceptance testing (see Fig. 8.3.19) and is derived from the need to ensure that the fabrication process is working as it is supposed to work. The concept came into being during metal bonding when adhesive qualification coupons were run along with bonded assemblies. As they were processed together, successful coupon tests were taken as evidence that the process was being adequately performed. This concept was extended to composites (tag-end tests are usually called out on composite engineering drawings; see Appendix B) with the following considerations:

(a) The test specimen must reflect the process (cure cycle) that is occurring at critical locations (often more than one on large complex parts). This means the significant factors in the process must be reflected in the fabrication of the test coupon.

(b) The coupon should be made along with the part (same material, same exposure to contamination in addition to the aforementioned cure cycle)

(c) The coupon should be a standardized type so that acceptance criteria can be established with statistical relevance

(d) The method should be simple enough to be systemized to the point that only manufacturing quality control interface is involved. The engineer should only become involved during set-up and when deviations to the requirements occur

Tests must satisfy items (a) and (b) above and should be conducted on pieces trimmed from production parts. However, this approach is contrary to items (c) and (d) requirements and therefore needs a trade-off study for the most cost effectivity.

Carpet Plots (Design Curves)

Tests are conducted on different laminates which contain varying percentages of $0°$, $\pm45°$ and $90°$ plies and which cover a range of layup configurations to be used for structural applications. Fig. 8.3.20 is a carpet plot showing the general relationship between strength and various layups of composite tape. Points Ⓐ, Ⓑ and Ⓒ are laminates used to define the lamina stiffness and Poisson's ratio properties of the material for predicting the strength and stiffness of various layups. Laminates Ⓐ, Ⓑ and Ⓒ are not used by themselves in structural applications.

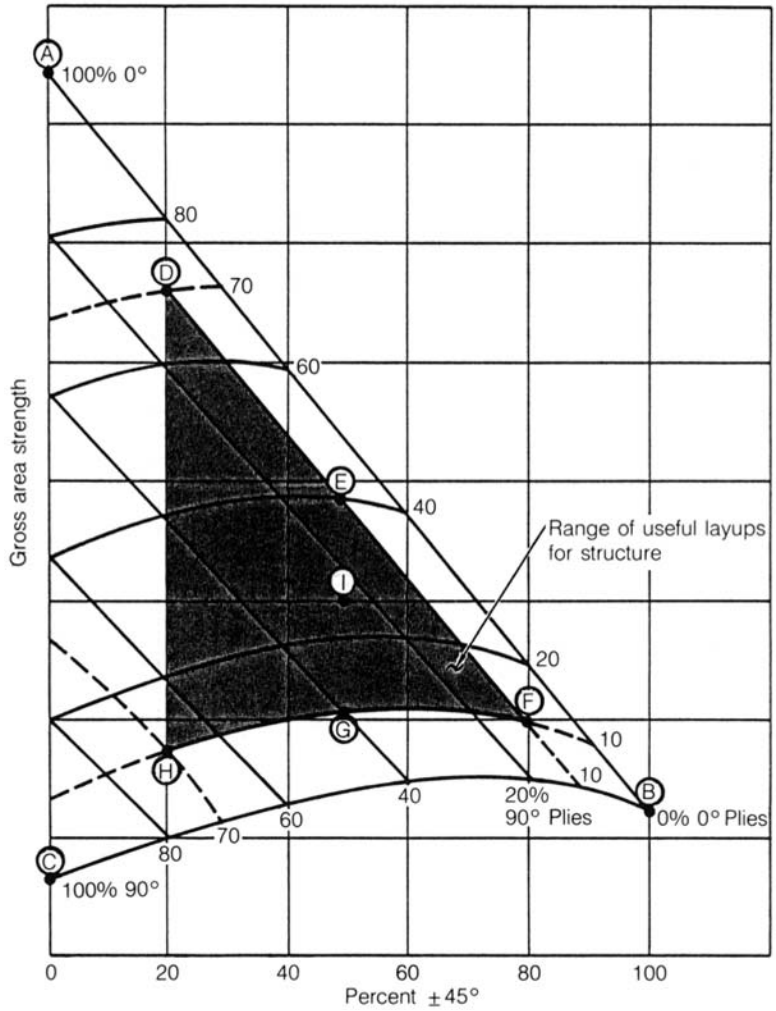


Fig. 8.3.20 Typical Carpet Plot For Family Of 0°, ±45°, 90° Tape Laminates (Example)

The crosshatched area indicates the range of laminates that are most useful for structural design applications. Laminates generally used in structures contain a minimum of 10% 0°, 10% 90°, and 20% ±45°, plies. Data for laminates (E) and (G), and (D) and (H) can be obtained from one laminate tested in two different directions. Laminate (I) is an isotropic laminate with equal amounts of material in four directions. Tests conducted on laminates (D) through (I) will be used to characterize the tape composite strength properties for the whole crosshatched region.

The properties for different percentages of 0° , ±45°, 90°, bidirectional fabric plies in laminates do not vary as much as do tape laminates. Each ply consists of fibers oriented at 0°/90° or at ±45°. An illustration of the variation in strength based on the percentage of 0°/90°, 90°/0° and ±45° plies is shown in Fig. 8.3.21. The strength is slightly higher in the 0° (warp) direction than in the 90° (fill) direction because the percentage of fibers is slightly higher in the 0° direction. The laminates (A), (B) and (C), shown in Fig. 8.3.21, are not used in structures, but the properties of these laminates are used for predicting the properties of other combinations of laminates.

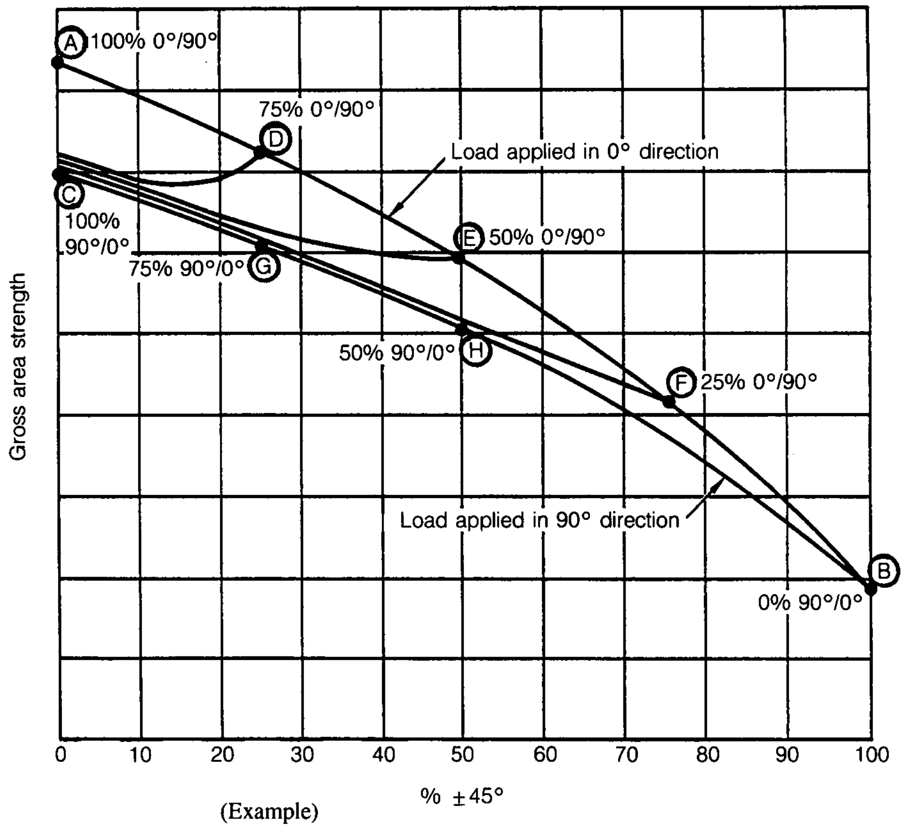


Fig. 8.3.21 Schematic Of Strength Characteristics For 0°/90°/±45° Bidirectional Fabric Laminates

AML Plots

This is a quick method of preliminary design allowable (also see Section 7.2 in Chapter 7). Since controversy exist over the use of this method, here, the text merely shows the procedures of creating AML plots. It can be obtained from the laminate-level coupon (3 specimens are used per batch and approximately 300 — 400 coupons are required) tests by the following steps:

(1) Laminate-level tests should conducted on the following configurations:
- (50/40/10) — i.e., % of plies (% of 0°/ % of ±45° /% of 90°)
- (25/50/25)
- (10/80/10)

These three configurations have AML's (percentage ±45° minus percentage 0° plies) of −10, 25 and 70, respectively.

(2) Unnotched tension and compression tests are performed on the above three laminate configurations at RTD (room temperature dry) and on the (25/50/25) laminate configuration at ETW (elevated temperature wet). These specimens contain measuring instruments which determine modulus as well as failure stress and strain.

(3) Specimens with an open hole are tested in both tension for OHT (open hole tension) and compression for OHC (open hole compression) at RTD, and a lower percentage are tested at ETW and CTD (cold temperature dry).

(4) Specimens with fasteners are tested for FHT (filled hole tension) at RTD with some at ETW and CTD.
(5) In addition to the tests of OHC and FHT described above, OHC and FHT specimens are machined from a (25/50/25) panel and tested at RTD to determine statistical factors.
(6) Specimens are impacted at specified "A" impact level and tested for compression at RTD and ETW.
(7) Quasi-isotropic (25/50/25) specimens are tested for PIC (post-impact compression) at RTD after impact at various specified impact levels.
(8) Specimens are tested in single-shear bearing using specified diameter (e.g. 0.25 inch) protruding head and/or countersunk fasteners to determine bearing strengths at RTD and at ETW.
(9) Pull-through tests are conducted using countersunk head fasteners (e.g. 0.25 inch).
(10) Picture frame shear tests are conducted on (10/80/10) laminates to determine initial buckling and the effect of specified "A" impact level [see item (6) above] on shear strength.

The test data (an example of test data for RTD is shown in Fig. 8.3.22) should be carefully reviewed and an appropriate "knock down" factor, e.g., 80% to 90% be determined.

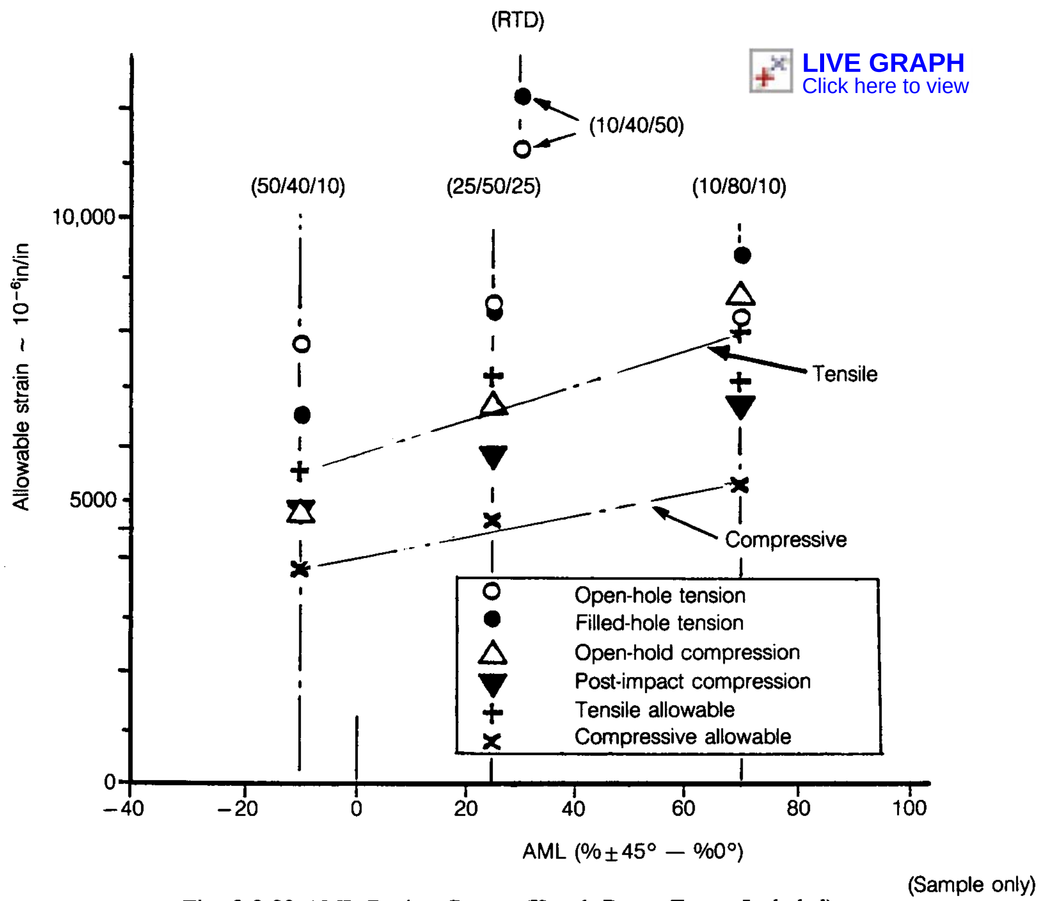


Fig. 8.3.22 AML Design Curves (Knock Down Factor Included)

8.4 ELEMENT AND COMPONENTS TESTS

The design of composite structures is often verified by testing actual structural components, as shown in Fig. 8.4.1 and Fig. 8.4.2; tests are used for allowable verification and for fulfillment of structural integrity requirements. These specimens occasionally contain holes, notches, stringer run-outs, joggles, etc., and typically have more instrumentation and require more effort to both load introduction and test fixtures, as shown in Figures. 8.4.3 through 8.4.5, because few standard methods are available at the component level.

Load-carrying structures usually contain sections which require intensive detailed analytical and experimental examination to ascertain their effects upon the performance of the total structure.

For example:

- An access hole through a skin structure may drastically alter the stress concentration and redistribution in the surrounding area
- A fastened, bonded and/or fastened joint likewise can produce significant stress perturbations in its immediate vicinity

These sections of components may induce large stress perturbations in the constitutive material and induce failure modes very different from those predicted by lamination theory.

In addition to inplane axial and shear loads, concentrated normal tension load on a composite integrally stiffened panel tests the flatwise tension and peel strength between skin and stiffener which are much lower than inplane laminate strengths. Thus, stiffener pull-off strength tests should be conducted. Fig. 8.4.6 shows a test in the form of a simple method which allows the pull-off results to translated into a running-load strength.

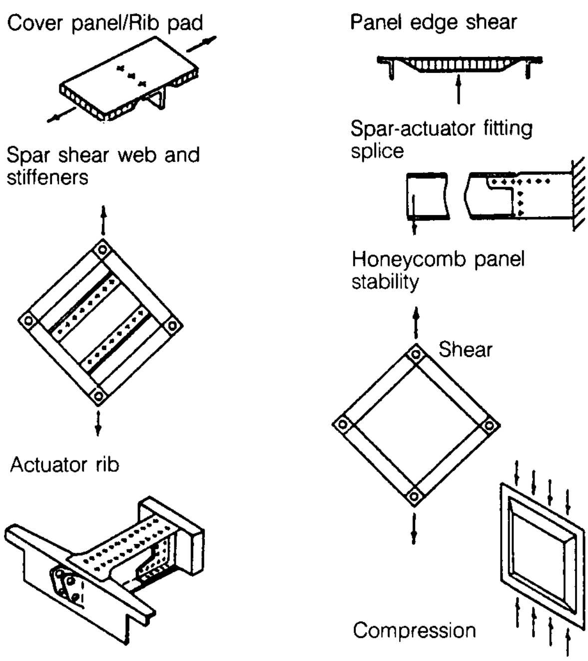


Fig. 8.4.1 Key Subcomponent Tests For B727 Composite Elevator

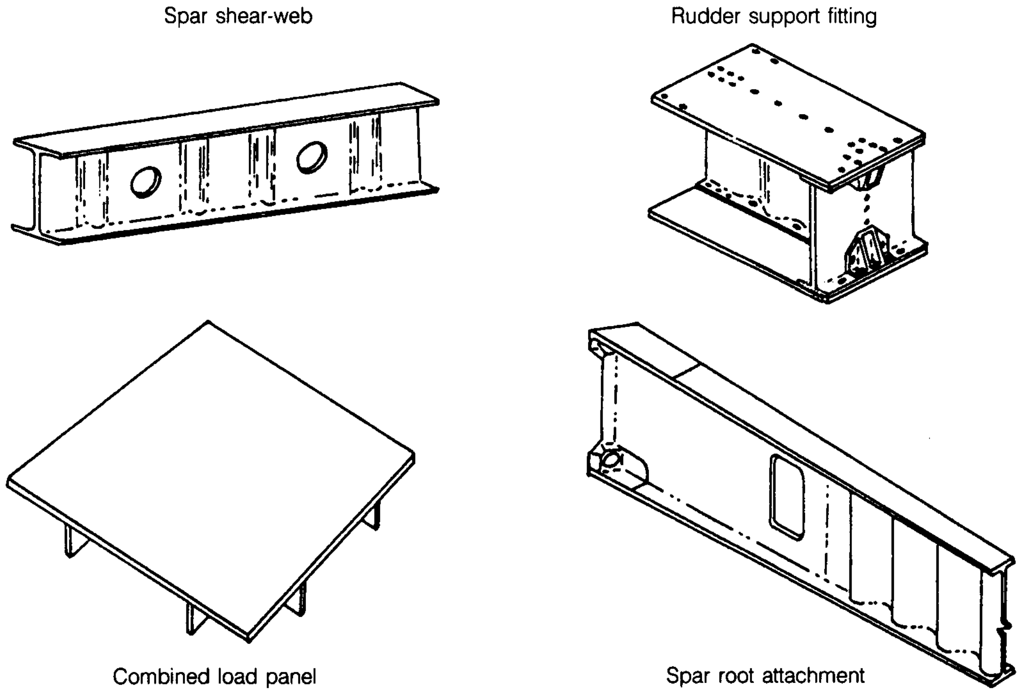


Fig. 8.4.2 Component Tests (DC-10 Vertical Composite Fin Box)

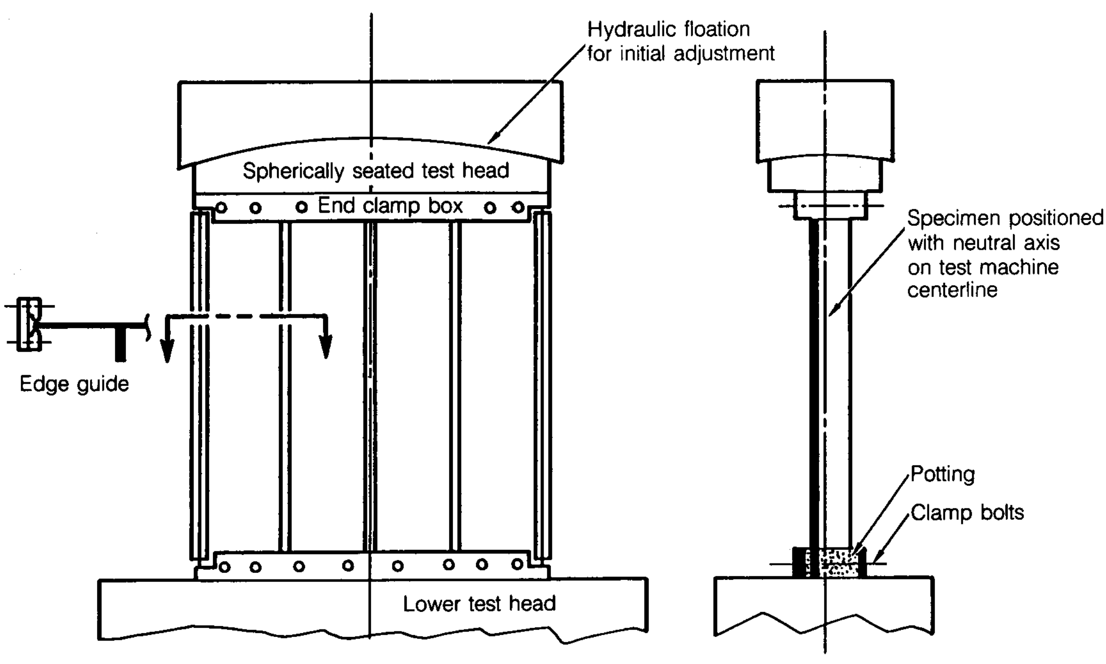


Fig. 8.4.3 Compression Test Panel Arrangement

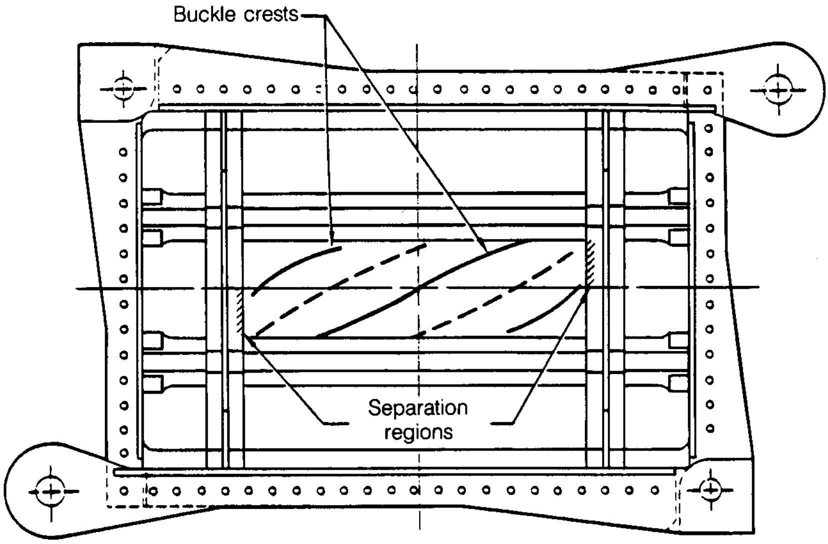
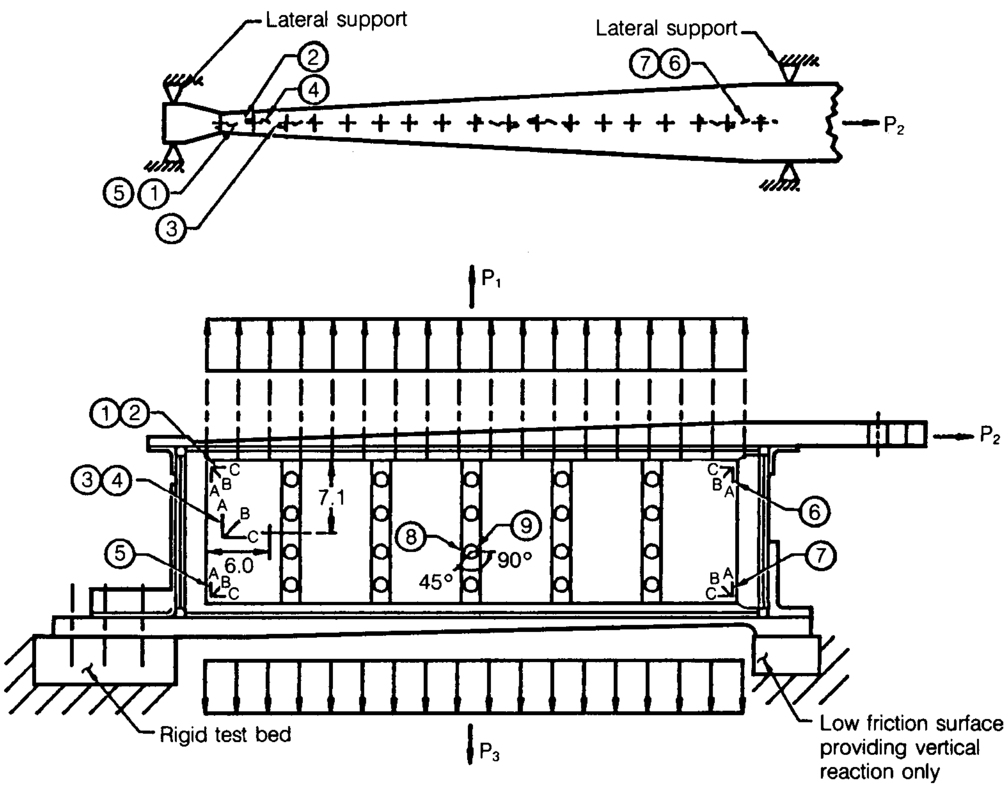


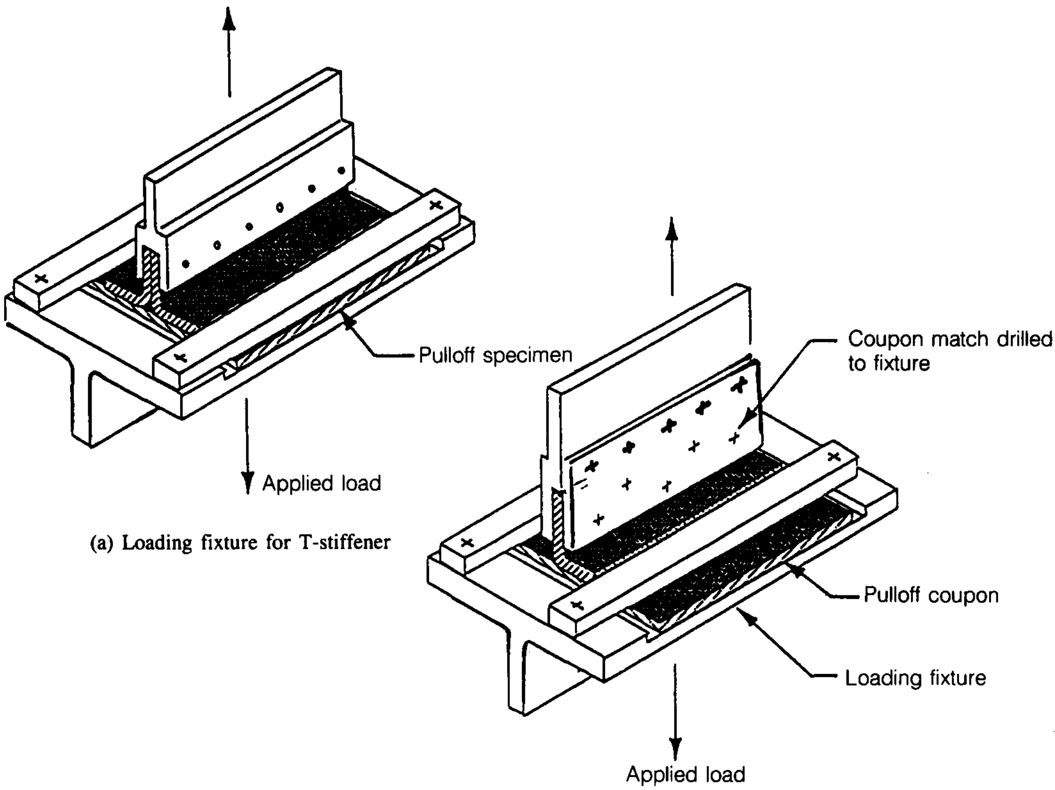
Fig. 8.4.4 Shear Panel Test Setup (Picture Frame)



Notes:

1. Strain gage rosettes ①, ②, ⑤, ⑥ and ⑦ are named in corners or web, reinforcement intersection.
2. Strain gage rosettes ③ and ④ are centrally located on list as shown.
3. Strain gages ⑧ and ⑨ are inside hole as shown.

Fig. 8.4.5 Sine Wave Spar Combined Shear And Tension Test Setup



(a) Loading fixture for T-stiffener

(b) Loading fixture for angle stiffener

Fig. 8.4.6 Stiffener Pull-Off Strength Tests

Joint Design Between Specimen and Fixture

One of the primary challenges related to joint testing is introducing load into the joint in a fashion which is representative of the boundary condition of a test component specimen. For example, it may be difficult or virtually impossible to determine, much less duplicate in a test, the stiffness boundary conditions which are present at the joint in actual service. The choice of boundary conditions which are readily reproducible in most tests are either free or fixed supports. The engineer may be able to furnish information as to the testing procedure and gripping hardware which would be most appropriate for approximating in situ conditions. The service stress distribution in the components which border on the joint must be predicted by the engineer. Then it is possible to approximate the same stress proportions by using boundary control techniques which are related to an active feedback signal from the component under test. Such a test may be expensive, but the application might be critical enough to warrant resorting to such a technique.

It is often necessary to measure the deflection of the joint as test loads are applied. Load and deflection data can be combined to provide a measure of the stiffness of the joint. Obviously, it is difficult to use a strain gage to measure the sliding motions which occur in most joints. A simple deflection sensor is an ideal transducer for measuring joint deflections.

Cutouts

Obviously, small coupon test specimens are inappropriate for experiments focusing on cutouts (holes) or other imperfections unless the flaw being tested is small compared to the specimen width; coupons can also be adversely affected by free edge stress effects. Thus, panel with major and minor dimensions close to that of the actual structure is logically the obvious choice for notch, cutout or imperfection tests.

In composite structure, a large cutout will present a significantly different stress redistribution around edge of the cutout and an array of strain gages may be adequate for quantifying the strain distribution. But the tips of cracks cause steeper strain gradients which are more appropriately measured by an optical procedure (e.g., Moire' or photoelastic coatings). Preventing local or general stability failure of the tested panel should be considered if the test is to measure material properties. An alternate method which can be used to prevent a general stability failure mode is to restrain lateral deformation by means of a reaction fixture.

Free Edge Effects

The delamination problem which is associated with free edges in cross-ply laminates is mentioned in Section 7.3 in Chapter 7. Free edge normal stresses will likely be more severe in laminates with cutouts because large stress concentration exist in the vicinity of cutouts. Measurements of through-the-thickness deformation should be made at the cutout edge since this may be the most relevant measurement to support analytical characterization studies. In addition, strain gages, displacement sensors and optical methods all have potential application for delamination strain characterization.

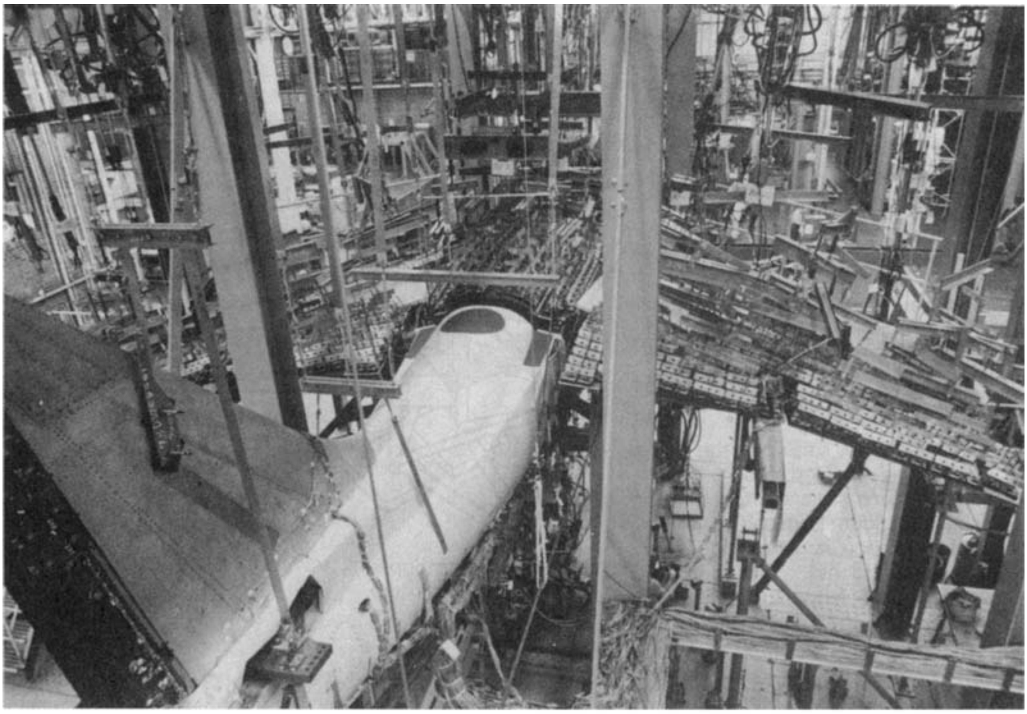
8.5 VERIFICATION FULL-SCALE TESTS

Full-scale testing (FST) of the completed airframe, as shown in Fig. 8.5.1 and Fig. 8.5.2, or testing of a large segment as a single unit (see Fig. 8.5.3), is the major test in an airframe structural test program. FST is one of the primary methods of demonstrating that an airframe can meet the structural performance requirements and it is extremely important because it tests all the related structures in the most realistic manner.

Typical FST includes static, durability (fatigue), and damage tolerance (may not be required in FST). The use of FST must take into account the unique characteristics of composite structures and their response to the expected service conditions as simulated by the test.

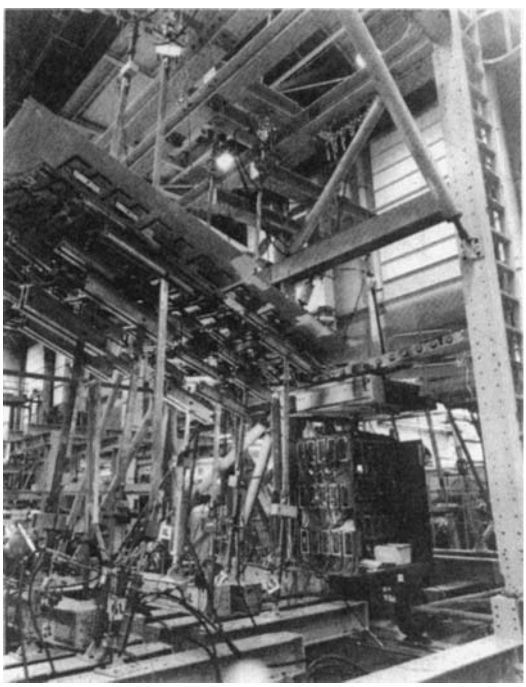
FST is a necessary check in the process of developing satisfactory structural systems. Analysis techniques have significantly improved in recent years, particularly with the advent of computers and the now possible finite element analysis; however, the complexity of composite structural systems still requires FST verification programs.

Test requirements such as limit and ultimate loads are often established on the basis of material test scatter derived from metals. Since composites usually exhibit higher scatter, problems may develop. Also, composite laminates exhibit relative brittleness, low interlaminar strength and a difference in CTE (in contact with metal parts) and all these factors present serious problems for the FST programs.



By courtesy of McDonnel-Douglas Corp.

Fig. 8.5.1 Full-Scale Test Of AV-8B VTOL Aircraft (Tension Patches Method)

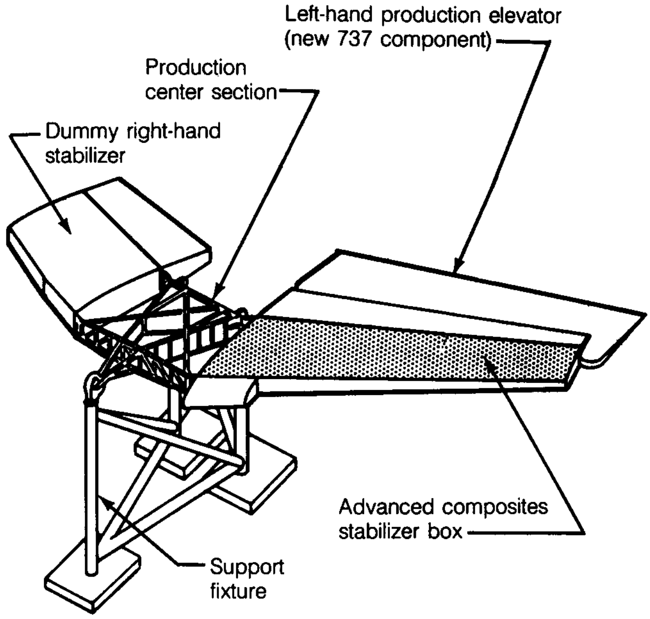


Full-scale ground test setup

Fig. 8.5.2 Full-Scale Test of B737 Composite Horizontal Stabilizer Box Structure (Ref. 8.17 And 8.18)

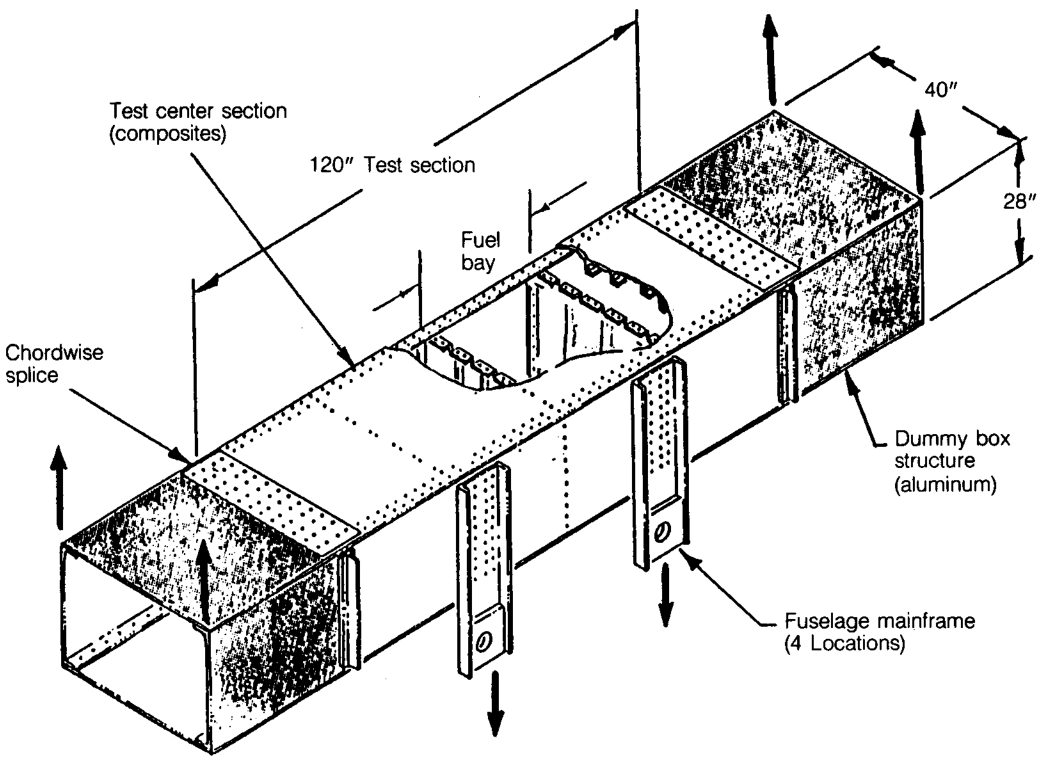
Test concluded:

- Limit load strain survey, four static load conditions
- Thermal linkage functional test
- Elevator attachment stiffness test
- One-half lifetime fatigue, no damage
- Damage tolerance, small damage
  - One lifetime repeated loads
  - Fail-safe, three static load conditions
- Ultimate load, four static load conditions
- Damage tolerance, large damage, four static load conditions
- Tip section subjected to lightning test



Full-scale ground test

Fig. 8.5.2 Full-Scale Test of B737 Composite Horizontal Stabilizer Box Structure (Ref. 8.17 And 8.18) (cont'd)



(Dimensions are in inches)

Fig. 8.5.3 Major Segment of A Composite Wing Box

There are three considerations when choosing the size of the test article:

- It must be large enough allows for proper complex loading and also for load interactions at interfaces that would otherwise be difficult to simulate
- If the component is small enough it is less costly to use a FST environmental test to certify the structure
- Structural configuration also play a role in the environmental condition test:
  - Primary or secondary structure
  - Type and complexity of loading

The purpose of FST is addressed below:

- To verify analysis with actual internal load distribution
- To see if any unexpected discrepancies occur
- To evaluate whether durability and damage tolerance have been adequately assessed
- To evaluate the durability of combination of composites and metals, particularly in interface areas (e.g., thermal expansion problems)

Instrumentation (all data is electrically recorded and controlled by computer) used on FST structures includes:

- Strain gages (most common)
- Deflection indicators
- Accelerometers
- Stress coatings
- Acoustic emission detectors
- Evener systems

Pre-test predictions of FST structure failure loads, locations and mechanisms are important to the test plan. These are based on minimum margin of safety calculations and the known statistical variations of the material allowable developed from coupon tests and used in analysis.

Appropriate "knock-down" factors are applied to the test margins after completion of the mechanical property (moisture) is testing. These results must be verified with a long-term aging study where the structure is subjected to real-life environmental conditions and tested at various intervals throughout the duration of the test program. Perfect duplication of temperature/moisture/time histories is not possible on complex FST structures and even attempting it can lead to unacceptably high test costs.

If FST must include environmental effects. It can be anticipated that FST will have to be wet-conditioned, damaged and tested in an environmental chamber having the capability of qualifying structures at the extreme temperatures specified for the design. The problem is workable and not as formidable as it sounds, but development work must to be done to accurately assess the costs and additional program risks associated with these requirements.

When composite mating structures are required for load introduction extra care must be taken to diffuse loaded into the structures. Generally:

(a) In tension tests — The mating structures must be sufficiently strong that they will not fail before the structure being tested.

(b) In compression tests — The mating structure must be simulated and the loads applied to it such that the rotational characteristics are approximated. This subjects components which are buckling critical to appropriate end-fixity conditions and ensures adequate load diffusion into the tested structure.

Static Test

The FST static test is the most important test in qualifying composite airframes because of their brittleness and sensitivity to stress concentrations compared to metal counterparts. The ability of the test data to meet certification requirements must be inherent in each of the FST static test requirements.

(1) The parameters to consider for the static test are:
- Type of test structure
- Type and number of load conditions
- Usage environment to be simulated
- Type and quantity of data to be obtained

It is difficult to conduct a FST under both temperature and moisture conditions but these environmental effects are addressed at the analysis, coupon, structural element, and component level (e.g., see Fig. 8.2.1) in the "building block" testing approach. The sums of these tests must be consolidated in such a way as to validate the consideration of environmental effects.

(2) The subject of loading methods on a FST composite structure needs careful consideration due to the composite's weak through-the-thickness strength (tension) and sensitivity to stress concentrations; possible test methods (e.g., on wing surfaces) are listed below:

(a) Tension-patches method (see Fig. 8.5.4):
- Offers uniform load distribution with closer representation of the real structural load (see Fig. 8.5.1); a more costly method
- Involves a more complex test set-up (higher cost)
- Introduction of load directly to the composite bonded surface must be done more carefully than with metal bonds because of through-the-thickness weakness

(b) Loading frame method (see Fig. 8.5.5):
- Less complex loading set-up and the least costly method
- All loads are converted into numerous compressive concentrated loads; this is not as effective as the tension-patches method but it is acceptable
- The attachment of substructures such as spars, ribs etc., at locations of concentrated loads needs careful investigation to make sure there is sufficient strength

(3) The recommended FST test sequence is as follows:

(a) Check the test set-up, which involves functional testing of:
- Loading jacks and evener system
- Instrumentation
- Data recording

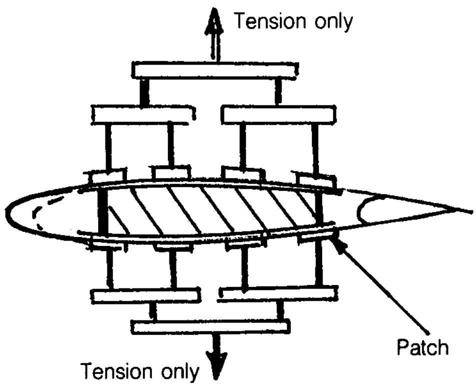
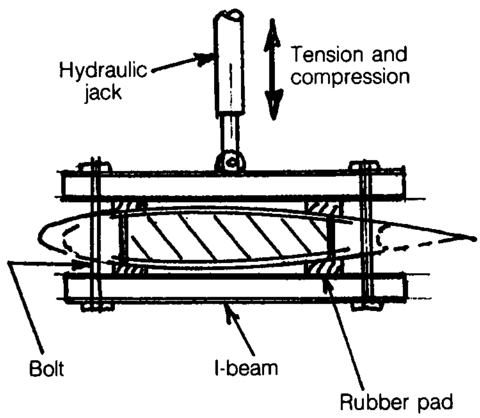
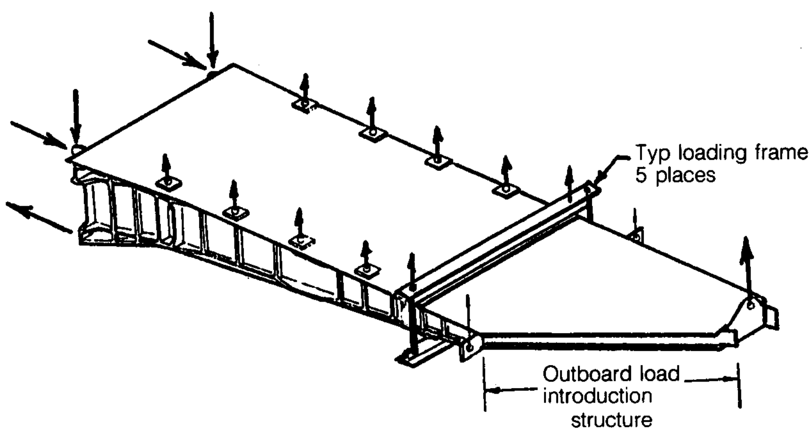


Fig. 8.5.4 Tension-Patches Method



(a) Loading frame arrangement



(b) Test setup example

Fig. 8.5.5 Loading Frame Method

- Real-time data displacements

This check can usually be done by applying a simple loading case at low levels to ensure that loads are introduced as expected.

(b) A strain and deflection survey is run to determine whether the strain distributions and deflections are as predicted. This is also done by applying low level loading (a simple load less than 50% of the design ultimate load) that will not affect the certification test results.

(c) The lowest of the loads to be certified are applied first:
- The conditions for which there is the highest confidence of success are run first
- Those conditions with the highest risk of premature failure are run last

(d) The early test results can be extrapolated to the predicted design ultimate load level

(e) If there is some risk of failure before reaching the required design load, a careful review and investigation should be conducted.

(4) Ultimate load requirement — Type of load required by the qualifying or certifying agencies to meet their validation requirements includes:

(a) U.S. FAA requires the structure to be tested to the limit load (same as conventional metal structures)

(b) U.S. military usually requires testing the ultimate load:

After completing the required FST, whether to limit or ultimate load, the airframe manufacturer usually chooses the most critical load condition to test to failure. If the destruction failure load exceeds the design ultimate load, the airframe future growth margin is warranted.

(5) The final step is a review of the data obtained from the test and evaluation of its correlation with the stress analysis.

Durability Test

Cyclic testing of structures used to evaluate metal structures is also used with composite structures. In general, FST cyclic testing is limited to 2 to 4 lifetimes of spectrum loading, including a spectrum load enhancement factor such as environmental effects. Periodic inspections must occur during FST durability testing at specified intervals. These inspections are conducted to determine whether any damage is progressing due to cyclic loading:

- To obtain the durability performance of the structural details
- To detect any critical damage whose growth would cause loss of the test article during testing

For example, stiffness change in a composite structure has been found to be an indication of fatigue damage; these inspecting for cracks and delaminations (very difficult to detect) is conducted at various times throughout the test. Nondestructive inspection methods with ultrasonic and x-ray are commonly used to detect damage.

Finally, a post-test inspection of the test article offer the FST durability test is very important to ensure that no damage has occured which would threaten the structural integrity of the composite design.

Damage Tolerance Tests

Testing composite FST structures for damage tolerance is especially important because it addresses the concerns associated with both the static and durability tests. The damage tolerance test, like the static test, is a qualification requirement of both civil (e.g., FAA) and military authorities. The load specified by civil and military requirements varies (both specify a residual strength requirement) and the requirements also vary depending on:

- Ability to inspect damage
- Type of service inspection used
- Type of aircraft

As in the durability test, the critical flaw or damage may be associated with either its initial state or its growth after cyclic loading. The environmental effect during the cyclic test is not easily defined but the load enhancement of the spectrum as recommended for the durability test is the best option. Because the FST damage tolerance test has many similarities to the static and durability tests, all the testing considerations which apply to them are also applicable to this test.

If the residual strength test is successfully passed ,the structure can then be loaded to failure to further evaluate or integrate its damage tolerance capability.

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Chapter 9.0

QUALITY ASSURANCE

9.1 INTRODUCTION

Non-destructive inspection (NDI) generally refers to the examination of a part or assembly in such a manner that the test article is not affected in any way. The methods used in determining composite quality include visual, Eddy-current, ultrasonic, radiographic (X-ray), and holographic inspection. With the small design margins used for airframe structures, NDI is essential to insure that composite structures are free from anomalies and can provide necessary performance.

The original impetus to develop a database on aging materials came from the nuclear power industry, which required more than 40 years of service life/safety from materials that would be subjected to critical wear and exposure during the life of a nuclear reactor. A materials aging database helps engineers to determine the proper materials to use for specific airframe applications. Once the material is chosen, it becomes the job of maintenance to ensure that the material will stand up to the stresses they are designed to withstand.

Composites are prone to manufacturing defects that compromise structural integrity and when they do occur they are generally not visible to the eye. While composites have been routinely inspected for a long time, two factors combine to increase the need for better methods of evaluating structural integrity:

- Composite materials are beginning to be used in structurally critical areas such as empennage stabilizers, fuselages, fighter aircraft wings, etc.
- The growing popularity of using composite materials has increased production rates and has forced manufacturers to look for smaller flaws and do it faster than ever before.

No matter how well a composite component is made, close examination inevitably reveals flaws; typically, large flaws in high-stressed regions are of the most concern. Deciding which flaws are critical, the size of flaw that can be ignored, and locations where such flaws can be tolerated or are unacceptable must be established in the design process. Acceptance criteria can vary widely depending on the application. An airframe primary structure obviously would have more stringent requirements than would a secondary structure.

Composite structures are difficult to design for visual inspection. Internal damage to a laminate skin is rarely apparent from the outside. Delamination or debonding of honeycomb core, caused by a dropped tool, frozen moisture or lightning, can be seen by X-ray; but because even a few square inches of internal damage can be unacceptable, a labor-intensive NDI scan of the whole component becomes necessary. Unlike the structure of most metallic materials, the built-up nature of composite laminates and honeycomb panels makes it easy for defects to occur. When a metal skin is hit it visibly dents, but when composites are hit nothing can be seen on the outside surface which unacceptable damage has occured to the internal structure.

It should be noted that, to keep controls adequate and cost-effective, many of the control functions cannot be the responsibility of Q. A. alone. Q. A. naturally retains the overall responsibility for structural product integrity, but certain areas of control must be delegated to Engineering, Planning, and Manufacturing. The task of quality control is to assure that composite parts represent a level of quality consistent with design requirements. This can be accomplished by planning, in cooperation with other disciplines, to develop the necessary documentation and specifications to facilitate the early detection of defects.

NDI is a test method which does not affect serviceability of the test article and its advantages are:

- Failure prevention
- Cost reduction
- Customer satisfaction
- Manufacturing process control
- Quality maintenance

In order to insure the proper quality assurance standard for a particular layup, the engineer must add a note to the drawing that calls out the controlling specification or document. Inspection criteria must stated and other requirement noted, such as those calling out NDI or testing requirements, on the composite engineering drawing.

There are three terms which generally need a full explanation to prevent misunderstanding:

- Non-destructive Testing (NDT) — This refers to the development and application of NDT methods and is the most general term
- Non-destructive Inspection (NDI) — This refers to the performance of inspections to established specifications and procedures using NDT methods to detect discrete anomalies
- Non-destructive Evaluation (NDE) — This refers to the capability to assess the state of a material, a component, or a structural form from a set of quantitative NDT measurements and to predict the serviceability of the item in question from these measurements when evaluated in the context of appropriate failure modes

Of these three terms this chapter is primarily concerned with non-destructive inspection (NDI) because the procedures and specifications which are implemented are used to detect defined anomalies, but in neither a quantitative nor serviceability predictive sense.

Function of Quality Assurance (Q.A.):

(a) Production quality analysis:
- Support team function with estimates (based on past experience) of defect and rejection rates (a percentage of parts inspected) for different types of manufacturing operations
- Recommend procedures for tracking part rejections and part variability during manufacturing
- Establish and implement procedures to identify the causes of high scrap/rejection rates or high variability in parts

(b) Quality assurance technology:
- Recommend Q. A./inspection procedures to aid Manufacturing in improving quality and reducing part variability
- Identify needed development of in-process inspection technology for error-avoidance and improvement of controlled-tolerance manufacturing
- Recommend procedures to integrate Q.A./inspection technology development with manufacturing technology and automation

In order to achieve an economical product which meets or exceeds requirements, it is essential that quality be considered as a goal throughout the design process. The engineer has the responsibility of designing a product which can be produced consistently with good quality and whatever tools are necessary must be employed to achieve that quality. Typical Q. A. involvement in the verification of a design is shown in Fig. 9.1.1.

Fabrication Level Inspection

(a) Fabrication inspection will include general surveillance to verify that:
- Manufacturing tooling has been conditionally accepted pending tool-try part acceptance
- Material and material utilization is controlled
- All measuring and test equipment calibrations are current
- Process specifications and engineering drawings are maintained in compliance with the manufacturing plan

(b) Incremental buy-offs during the fabrication process will verify:
- Tool preparation
- Number of plies
- Ply orientations after layup
- Consistency of ply trimming operations
- Proper layup in the tool

(c) Tool assembly, and bagging and leak check prior to cure will also be verified.

(d) During the cure, pressure, vacuum, temperature, and dwell time will be monitored, with a final buy-off recorded on the process control chart

(e) After cure, visual, dimensional, and non-destructive inspections will be performed in accordance with the engineering drawing and process specifications.

(f) Any non-conformances will be identified and the item will be held for engineering disposition.

(g) Process control specimens will be evaluated by the Q. A. laboratory and the results documented.

(h) The flow diagram shown in Fig. 9.1.2 gives the typical incremental inspections to be performed.

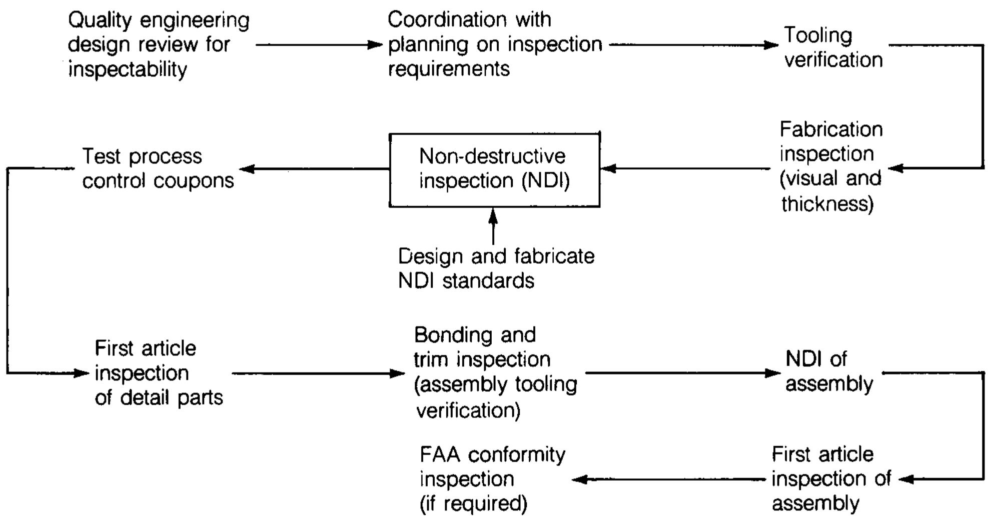


Fig. 9.1.1 Quality Assurance Involvement in the Verification of The Design

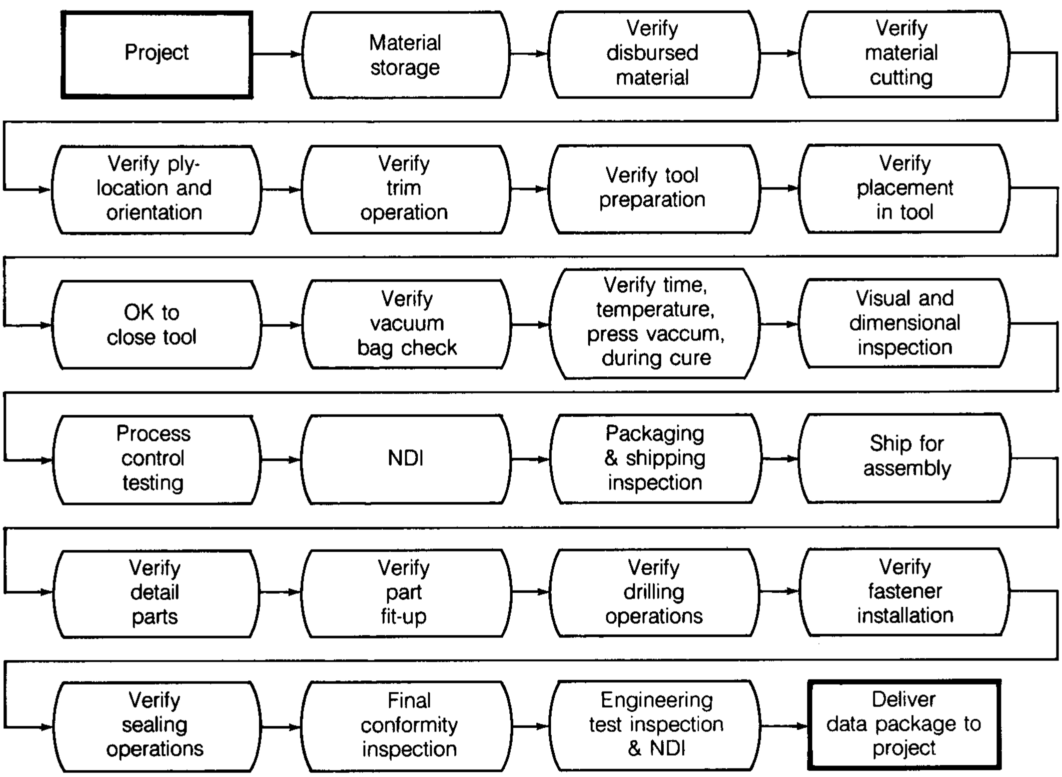


Fig. 9.1.2 Inspection Operations For Composite Fabrication

9.2 MATERIAL QUALIFICATION

Unlike the metallurgy industry, manufacturers of composite materials generally are unwilling to let airframe manufacturers know the chemical compositions of their materials. This has forced many airframe manufactures to hire their own chemists to analyze the materials received from the materials companies to better understand their chemical properties. This is further complicated by the fact that most airframe manufacturers will not exchange the chemical information they have gained, and such work is repeatedly duplicated.

Material Batch Acceptance

Upon material receipt, the receiving inspector verifies the shipment against the vendor's packing slip and checks the certification for that receipt. All inspections are conducted in accordance with the inspection codes that are indicated on the purchase order. A portion (a strip taken across the full width of the material) of each batch of a material is submitted to the engineering laboratory as a part of the receiving inspection acceptance procedure. Tests are conducted in accordance with the relevant specification called out on the purchase order. Two types of tests may conducted:

(a) Uncured properties tests — These tests generally consist of evalutating
- Ply thickness
- Tack (to determine the stickiness of thermoset materials)
- Volatile content
- Resin content
- Areal weight
- Moisture absorption (optional)

(b) Cured properties tests — These tests are conducted on a laminate fabricated from the specific batch of material received. The laminate is ultrasonically inspected prior to the laboratory tag-end test (see later discussion). The panel may be machined into coupons from which strength tests are performed. Cured laminate (ultrasonic inspected prior to test) properties tests are conducted on the specific batch of material received:
- Resin/fiber content — determines ratio of resin weight to fiber weight (results reported as a percentage)
- Void content — determines ratio of voids to resin and fiber weight for a specific volume (results reported as a percentage)
- Specific gravity — ratio of resin and fiber weight to the weight of water (reported as a ratio)
- Grind-down or burn-off — used to determine layup (results indicate the number and orientation of each ply)

A material safety data sheet (MSDS) must contain the following information (also see Ref. 1.2):
- Hazardous ingredients
- Physical data
- Fire and explosion hazard data
- Health hazard data

- Reactivity data
- Spill or leak procedures
- Special protection requirements
- Special precautions
- Method of transportation

Tag-end Tests

A Typical tag-end testing includes the following:

- Grind-down — ply orientation must conform to drawing
- resin content — by weight (e.g., $32 \pm 2\%$)
- Fiber volume — for information and reference only
- Void volume — for information and reference only
- Thickness — for information and reference only
- Laminate density — by lb/in^3 (or g/cc)
- Micrographs perpendicular to 0° — ply stacking sequence, porosity, and delamination
- Micrographs perpendicular to 90° — ply stacking sequence, porosity, and delamination

The tag-end example shown in Fig. 9.2.1 can be used to the above grind-down process listed above.

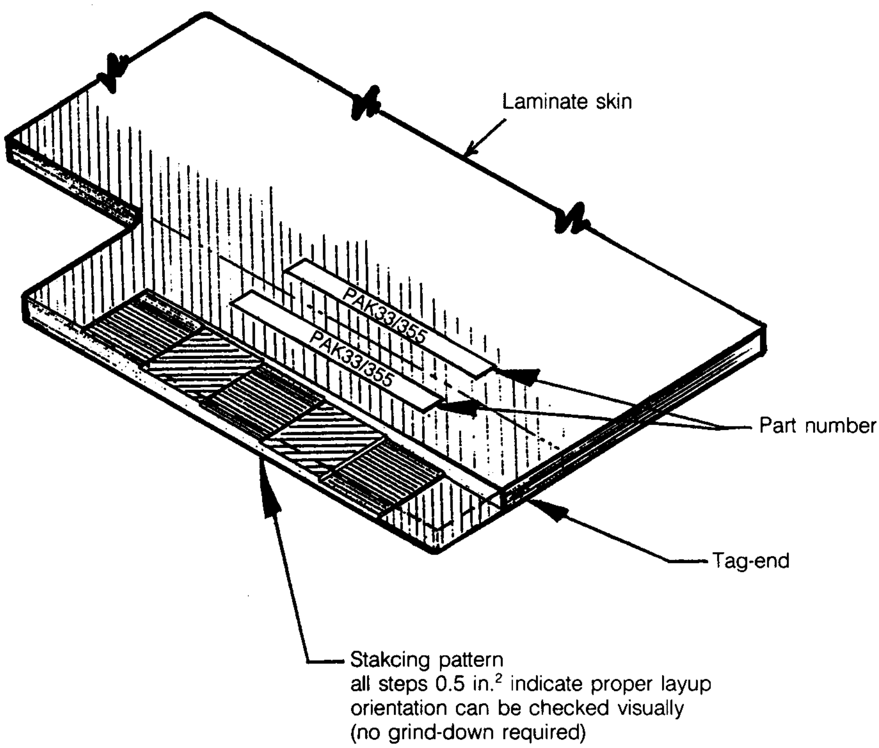


Fig. 9.2.1 Use Stacked Pattern To Avoid Grind-down Process

9.3 TYPES OF DEFECTS (FLAWS)

In composites, the many variables in the manufacturing process make many types of defects or flaws possible. Fortunately, most flaws occur infrequently, are easily prevented, or are not particularly detrimental to product performance. Defects that generally are of most concern are delamination and voids. These are gaps in the structure that, depending on location, can dramatically reduce strength.

(a) Delamination — Separation of plies within a laminate as a result of gas pockets or contamination. Delamination is caused by improper surface preparation, inclusion of foreign matter, or impact damage during shipping or handling.(see Fig. 9.3.1)

(b) Voids — Small clusters of air or gas micro-bubbles which have a tendency to collect at plies. Voids (see Fig. 9.3.2) are usually due to one of two factors:

- A mismatch in tooling during the cure cycle which results in unequal pressure distribution through the part, forming gaps that resin cannot fill.
- Resin with high volatile content combined with a short cure cycle. If volatiles cannot escape before the resin sets, voids will form. Occasionally, voids form when moisture is absorbed by the resin. Proper storage and process control will prevent this.

(c) Porosity — A condition of air or gas micro-bubbles in a given area within solid material. Caused by the incomplete flow of resin during cure and localized excessive heating or resin contamination (see Fig. 9.3.3). Mechanical properties do degrade relative to the severity of the porosity, but it has minimal effect if porosity is less than 3%.

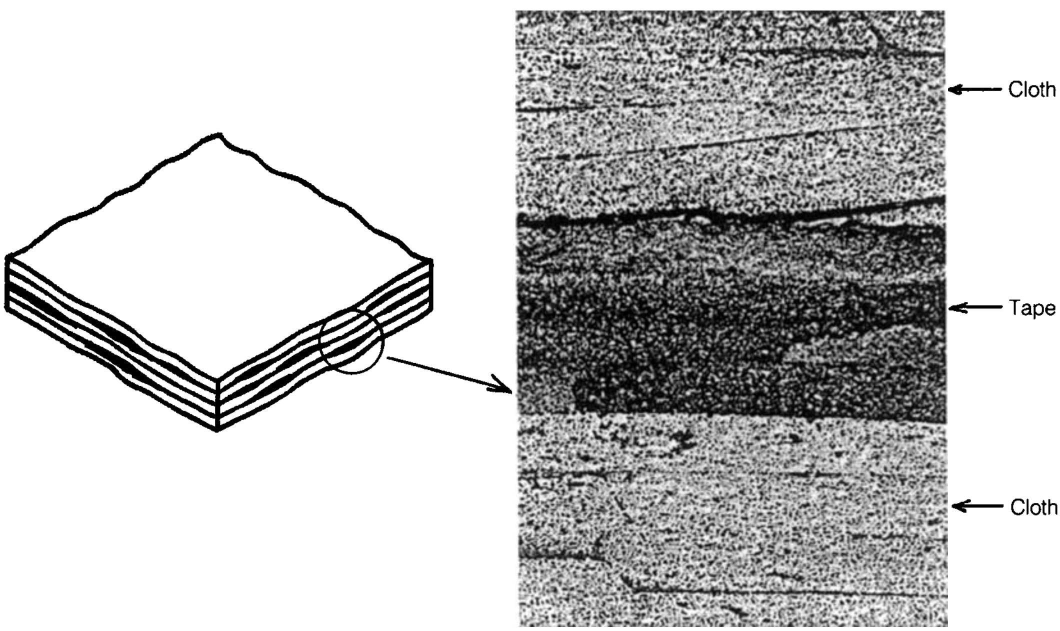


Fig. 9.3.1 Delamination

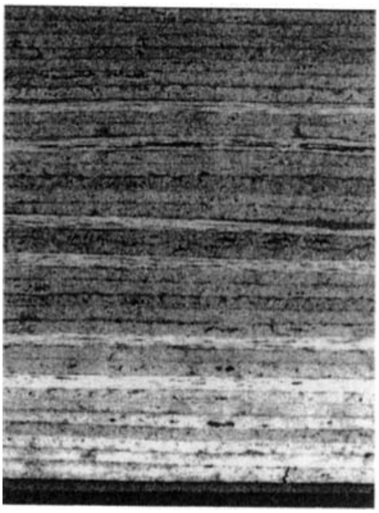
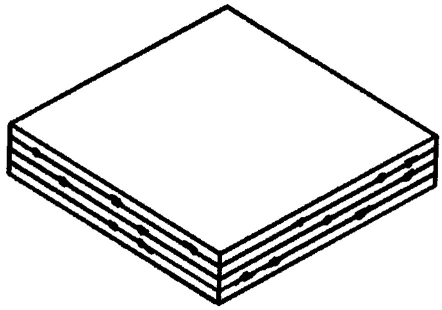
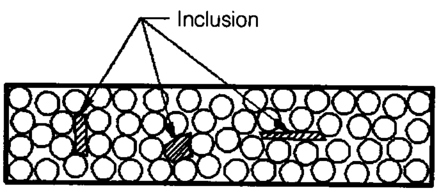


Fig. 9.3.2 Voids and Porosity



(See photo in Fig. 9.3.2 and Fig. 9.3.6)

Fig. 9.3.3 Porosity



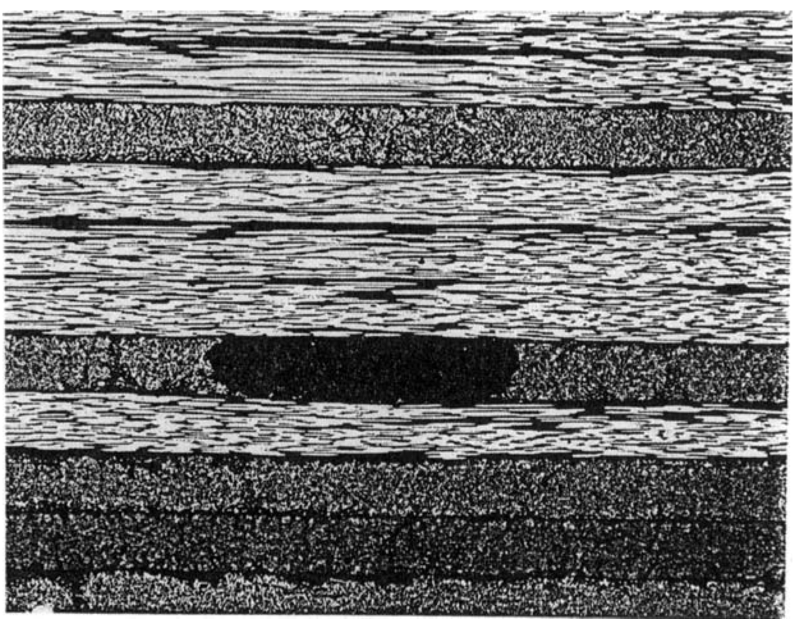
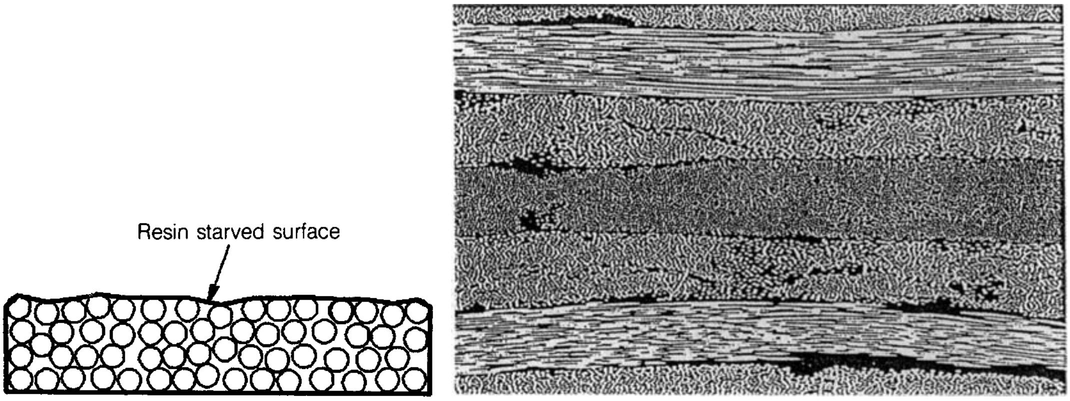
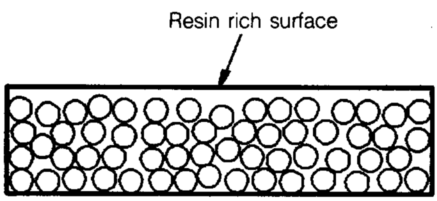


Fig. 9.3.4 Inclusions

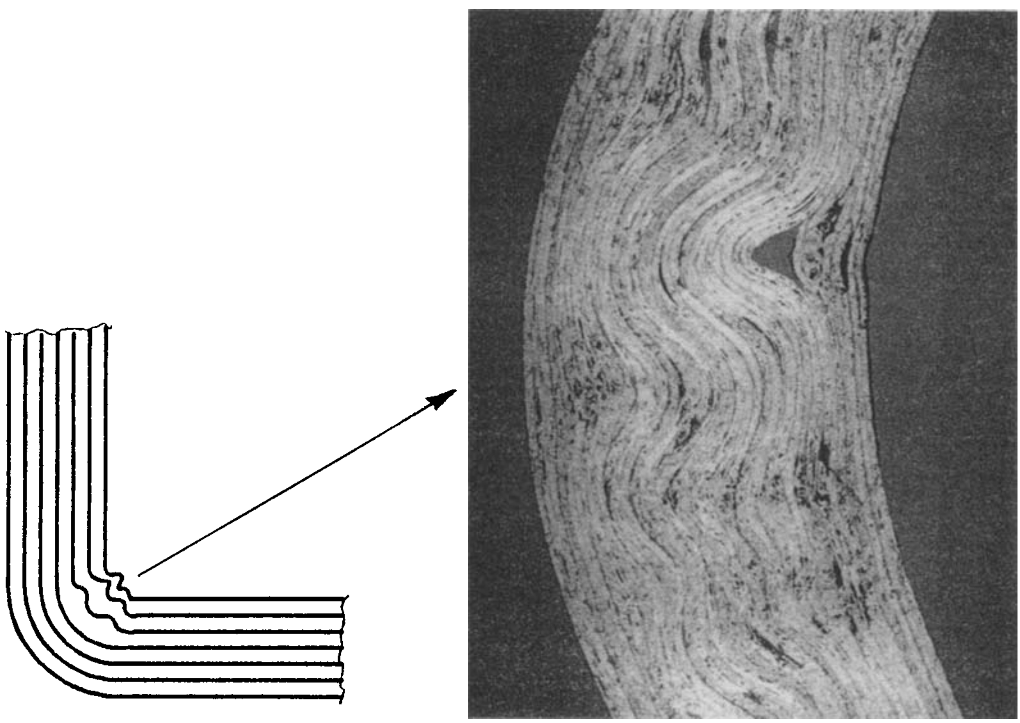


(a) Resin starved



(b) Resin rich

Fig. 9.3.5 Resin Variations



(Voids, porosity and fiber buckle)

Fig. 9.3.6 Fiber Variations

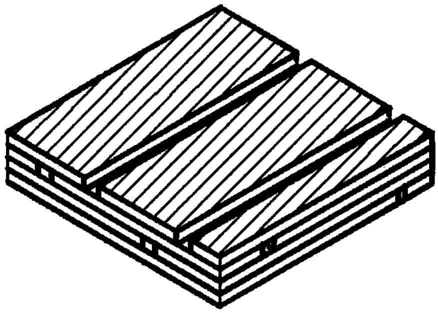


Fig. 9.3.7 Gaps at Splices

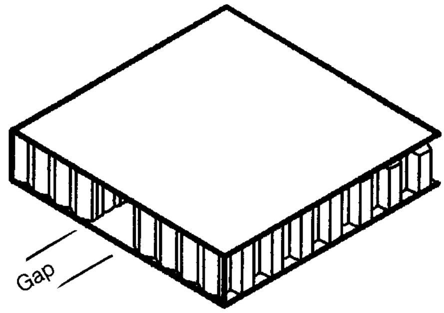


Fig. 9.3.8 Honeycomb Splice Gaps

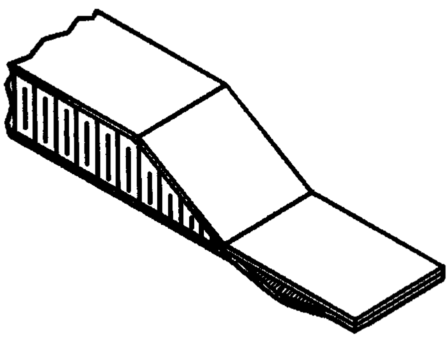


Fig. 9.3.9 Disbond

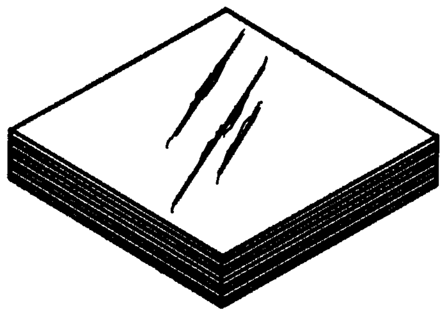


Fig. 9.3.10 Surface Scratch

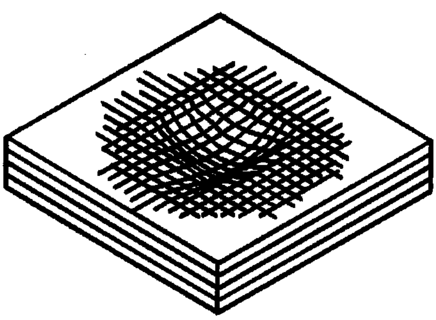


Fig. 9.3.11 Surface Depression

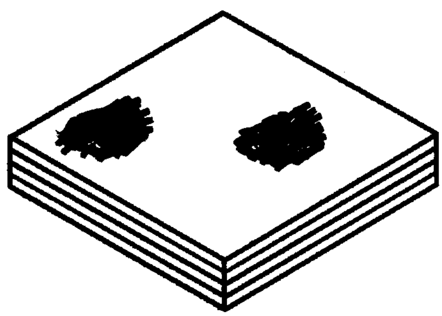


Fig. 9.3.12 Frayed Or Burned Material

(d) Inclusions — Foreign material inadvertently embedded in a layup such as chips, backing paper, tape, and other contaminants (see Fig. 9.3.4).
(e) Resin variations — Resin-rich or resin-starved areas may occur when laminate is improperly compacted or bleeding is improperly controlled during cure (see Fig. 9.3.5).
(f) Fiber variations — Abrupt changes in fiber orientation appearing as wrinkles, distortion, gaps, etc., resulting from problems in layup or cure (see Fig. 9.3.6)

(g) Gaps at splices — Gaps between tapes or fabric that are caused by poor layup practices (see Fig. 9.3.7).

(h) Honeycomb splice gaps — Open spaces between adjoining core or between core and face sheet are caused by improper layup (see Fig. 9.3.8).

(i) Disbond — Lack of bond joining between two different details because of surface contamination; bad fit between detail parts (see Fig. 9.3.9).

(j) Surface scratch — A narrow depression caused by marking or tearing with something rough or pointed; fibers may be broken (see Fig. 9.3.10).

(k) Surface depression — An indention caused by tools, or foreign material, resulting in deformed, not broken fibers (see Fig. 9.3.11)

(l) Frayed or burned material — Caused by extreme heat during cure and marked by noticeable color change in material (see Fig. 9.12)

(m) Broken fibers — Discontinuous or misplaced fibers in laminate (see Fig. 9.3.13)

(n) Incorrect ply count — Too many or too few plies in laminate, may not affect laminate thickness (see Fig. 9.3.14).

(o) Incorrect ply stack-up — Error in order of stacked plies in laminate (see Fig. 9.3.15).

(p) Incorrect ply alignment or orientation — In a composite laminate, alignment can typically vary by $\pm 2°$ in either direction without noticeable effect on overall strength. One problem that occurs occasionally is that plies are totally out of specified alignment, e.g., 45° or 90° is used where 0° is called for (see Fig. 9.3.16).

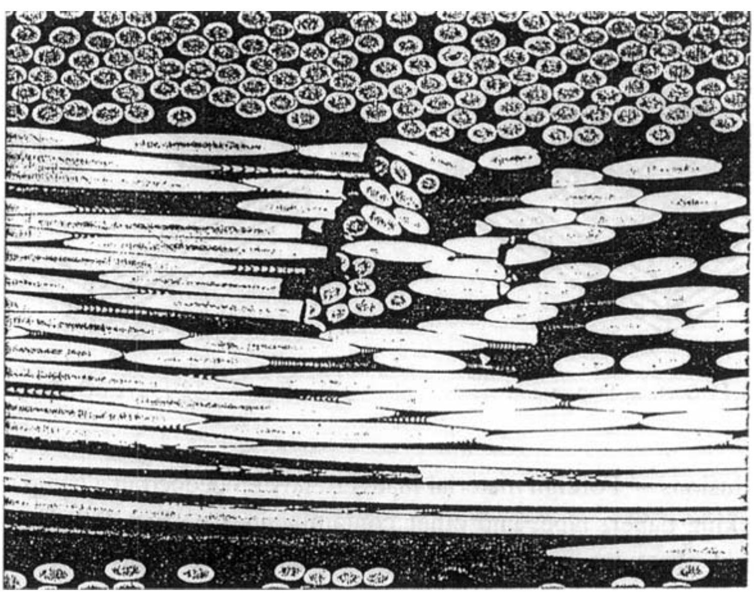


Fig. 9.3.13 Broken Fibers

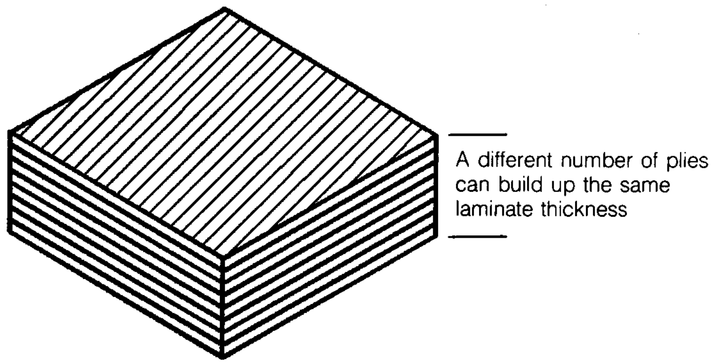


Fig. 9.3.14 Incorrect Ply Count

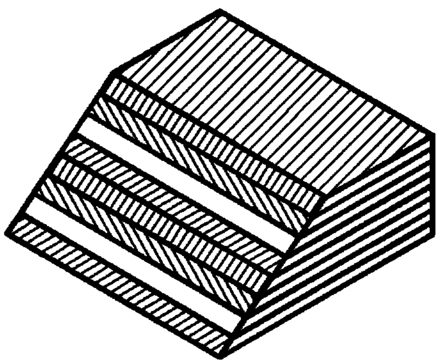


Fig. 9.3.15 Incorrect Ply Stack-up Sequence

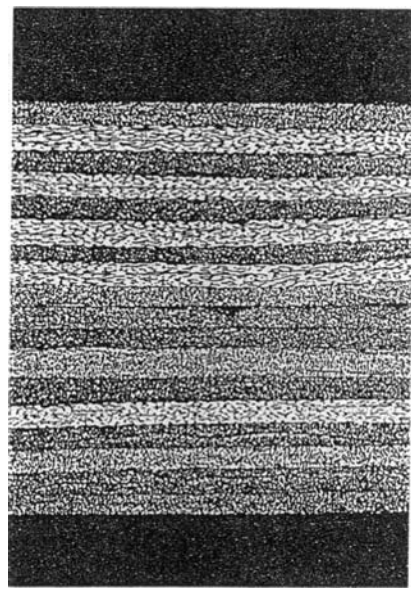


Fig. 9.3.16 Incorrect Ply Alignment

9.4 NDI METHODS

Visual Inspection

Visual inspection is the simplest and most economical inspection method and is routinely used for process control and final part inspection. Adequate, controlled lighting is necessary, and visual aids, such as mirrors, borescopes, magnifying glasses, microscopes, and other optical devices are used to inspect for defects and missing components. Fig. 9.4.1 illustrates the recommended angle to be used when visually inspecting with a light.

Visual inspection may be summarized as follows:

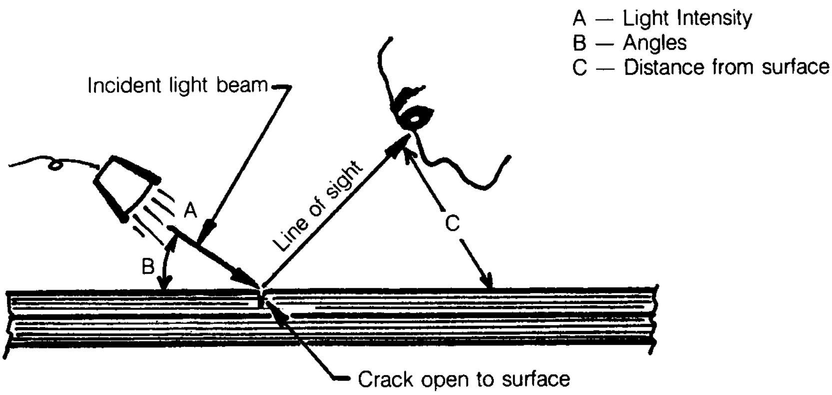


Fig. 9.4.1 Visual Inspection

- Requires adequate controlled lighting
- Utilizes visual aids (mirrors, borescopes, magnifying glasses, microscopes, etc.)
- Simplest and most economical method
- Routinely used for process control and final inspection

Ultrasonic Inspection

Ultrasonic inspection has become the most widely used method for detecting internal flaws in composite laminates and honeycomb assemblies. In this method, high frequency sound energy is introduced into the test part, and interpretation of the returned signals determines the presence of porosity, voids, delaminations, disdbonds, and other anomalies associated with composite materials (see Fig. 9.4.2).

Ultrasonic process:

- Methods:
  - Pulse-echo
  - Through-transmission
  - Reflected through-transmission
- Procedures:
  - Immersed (in water)
  - Contact (transducer on part)
  - Squirter (uses water stream)
- Displays:
  - A-scan (on CRT scope)
  - B-scan (on CRT scope and printout on x-Z recorder)
  - C-scan (on CRT scope and printout on x-y recorder)

Ultrasonic inspection:

- Relatively low cost inspection
- Is capable of providing permanent record for data retention in the form of hard copies and/or magnetic media

- High sensitivity to:
  - delaminations
  - voids
  - porosity
- Radiography complements the ultrasonic methods

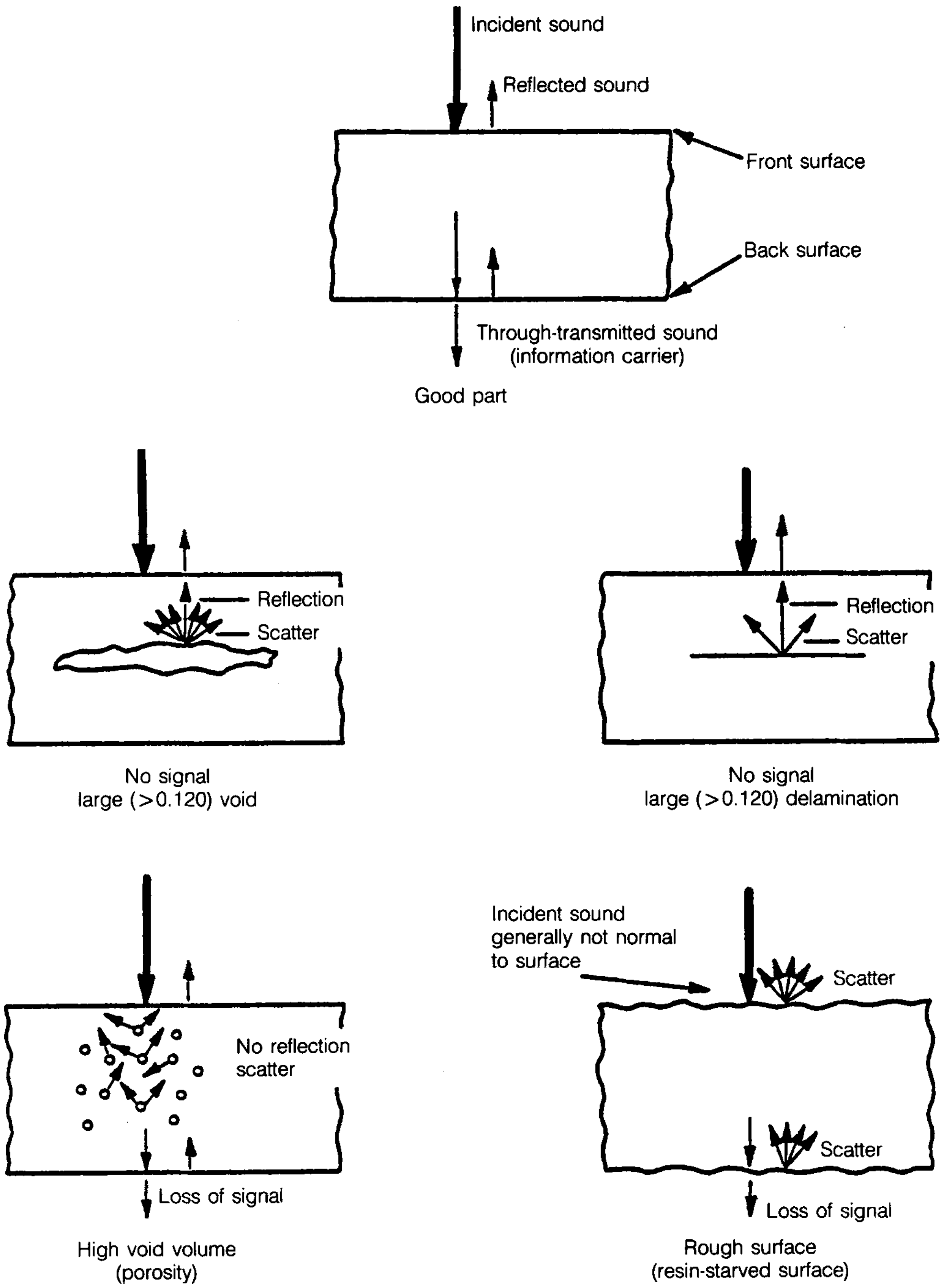


Fig. 9.4.2 Ultrasonic Wave Transmission Through Composite Laminate Panel

Disadvantages are:

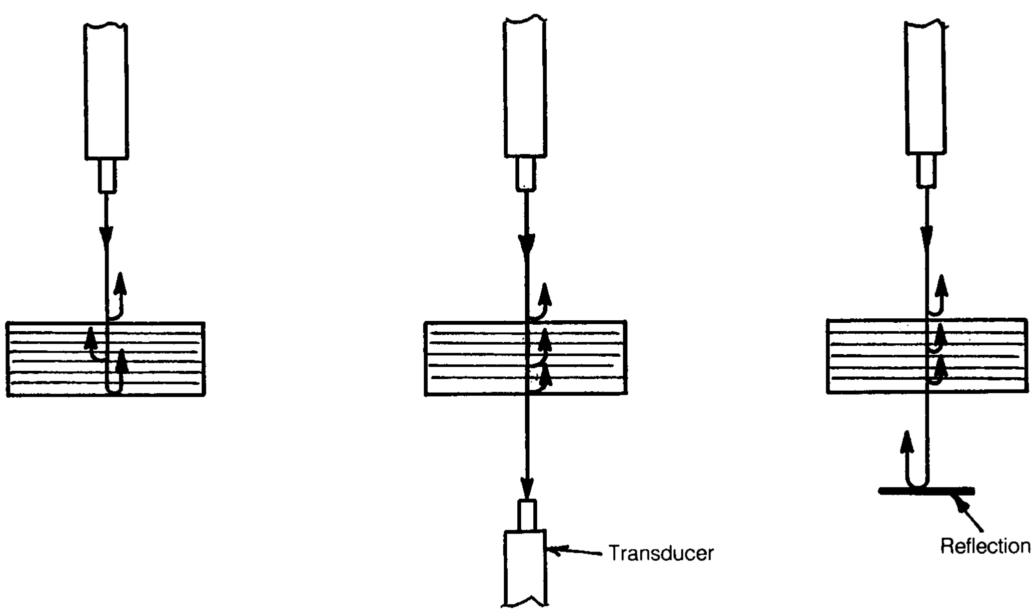
- Relatively slow scanning method
- Difficulty imaging complex shapes
- Unbonded surfaces in intimate contact (''kissing'') are not normally detectable (difficult to detect but can be detected)

Ultrasonic inspection is one of the few techniques capable of detecting flaws deep within the interior of a laminate. It is more sensitive to variations in organic materials, such as those used in most composites, which often do not absorb X-rays sufficiently to produce a high contrast. Whereas X-rays can completely miss delaminations and other thin, large area flaws that are perpendicular to the beam, ultrasonic inspection detects such flaws exceptionally well. An image is made of the test part by making time exposures on the screen as the part is being scanned.

Basically, three ultrasonic procedures, as shown in Fig. 9.4.3, are used for inspection of composites:

(1) Pulse-echo:

In the use of this procedure, ultrasonic energy is transmitted and received by one transducer. This energy is displayed on the ultrasonic instrument by signals which relate time and distance with amplitude (reflected energy). Energy is transmitted into the test part and the sonic energy is reflected from internal reflective surfaces. If no significant reflectors are encountered, the energy strikes the back surfaces of the test part, is reflected back to the transducer, and the received information is displayed on the ultrasonic scope. Thus, when properly adjusted, the scope displays the front surface signal, a back surface signal, and signals from any conditions between the front and back surfaces (see Fig. 9.4.4).



(a) Pulse/echo (b) Through-transimission (c) Reflected through-transmission

Fig. 9.4.3 Ultrasonic Procedures

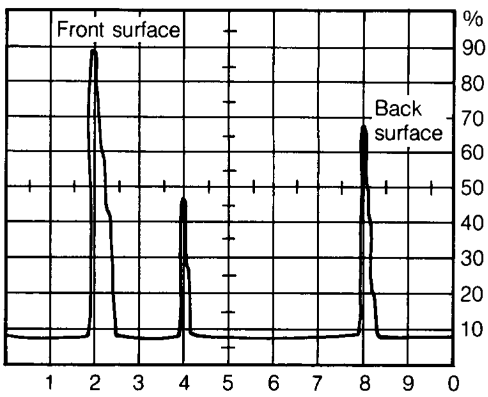


Fig. 9.4.4 Ultrasonic Scope Face

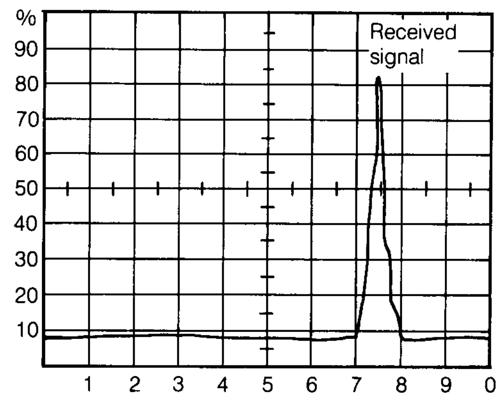


Fig. 9.4.5 Scope Showing Through-Transmission Signal

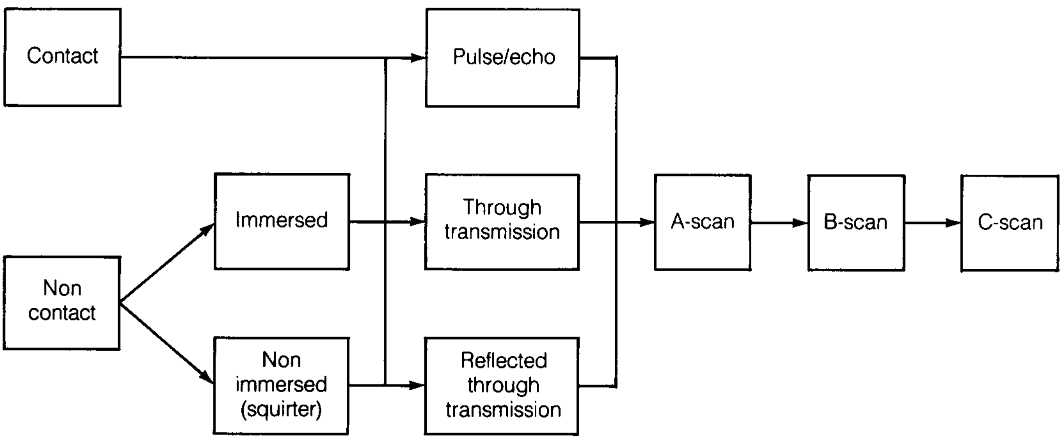


Fig. 9.4.6 Ultrasonic Inspection Procedures

(2) Through-transmission:

As the name suggests, ultrasonic energy is transmitted by one transducer and received by another, usually located on the opposite side of the test part. The ultrasonic scope displays a signal from the receiver as shown in Fig. 9.4.5. No reflections from the part interior are seen; the scope displays only the energy that passes through the part. If there is no flaw within the test part, the signal would remain at the calibrated height. If some anomaly interrupted the sound beam, then the received signal would be reduced. Honeycomb structures can only be inspected by through-transmission method (some pulse-echo can be done on honeycomb structure but is generally limited to laminate skin).

(3) Reflected-through-transmission:

This method is same as that of through-transmission except using reflector on the opposite side of the test part instead of transducer.

These procedures can be implemented in two ways (see Fig. 9.4.6):

(a) Immersed inspection — This procedure, as shown in Fig. 9.4.7, requires some type of medium, usually water, to transmit ultrasonic energy from the transducer to the test part. This is most frequently done by submerging the test part in a water-filled tank; a column of water (squirter) can be used, as shown in Fig. 9.4.8, but there must be no bubbles in the stream between the transducer and the part, as this will cause false indications of defects. The transducer, mounted on a moveable bridge scanning system, passes over the entire part. Either pulse-echo or through-transmission may be used and the information is displayed on a scope.

(b) Contact inspection — In this method, as the name implies, the transducer is in direct contact with the surface of the test part. The transducer moves over the surface to be inspected, as shown in Fig. 9.4.9. The transducer is coupled to the part by a thin non-toxic and non-corrosive sound conducting film, such as gelled water (tap water may be used as well). Either pulse-echo or through-transmission may be used, depending on the size of the part and accessibility of the inspected surface.

Two presentation methods are generally used for ultrasonic inspection of composite structures:

(a) A-scan:
A-scan is the scope presentation in which time or distance (on the scope baseline) is related to amplitude (scope height of the signal). Signals returned from the test part are displayed, whether contact or immersed, pulse-echo or through-transmission. The scope presentation is called "A-scan" and a typical display is shown in Fig. 9.4.10(a)(b).

(b) B-scan:
B-scan can also be used but is not a frequently used method. It produces a cross-section through-the-thickness image of the part.

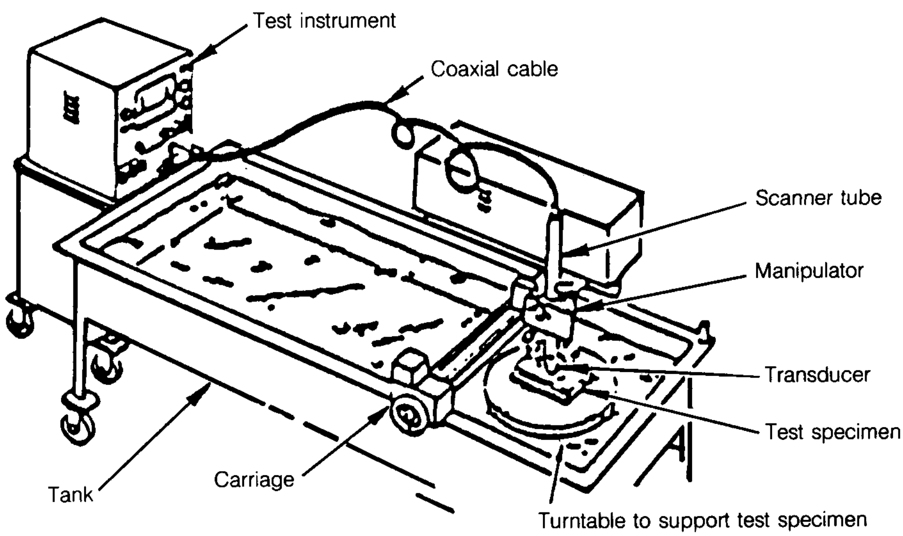
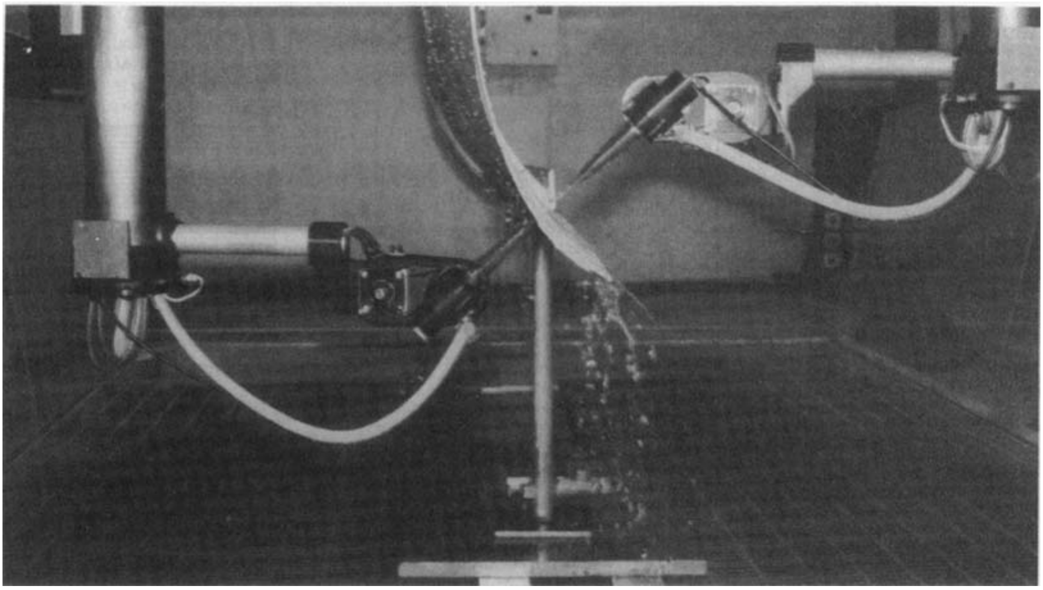


Fig. 9.4.7 Tank and Instrumentation for Immersed Testing



By courtesy of McDonnell Douglas Corp.

Fig. 9.4.8 McDonnell Aircraft AUSS-V System To Scan Contoured Composite Panels Inspected Using A Squirter

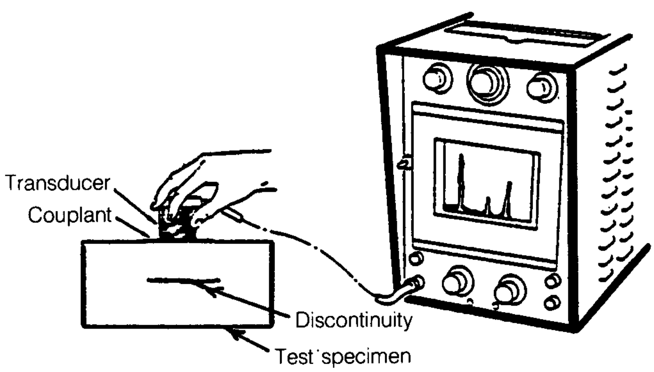
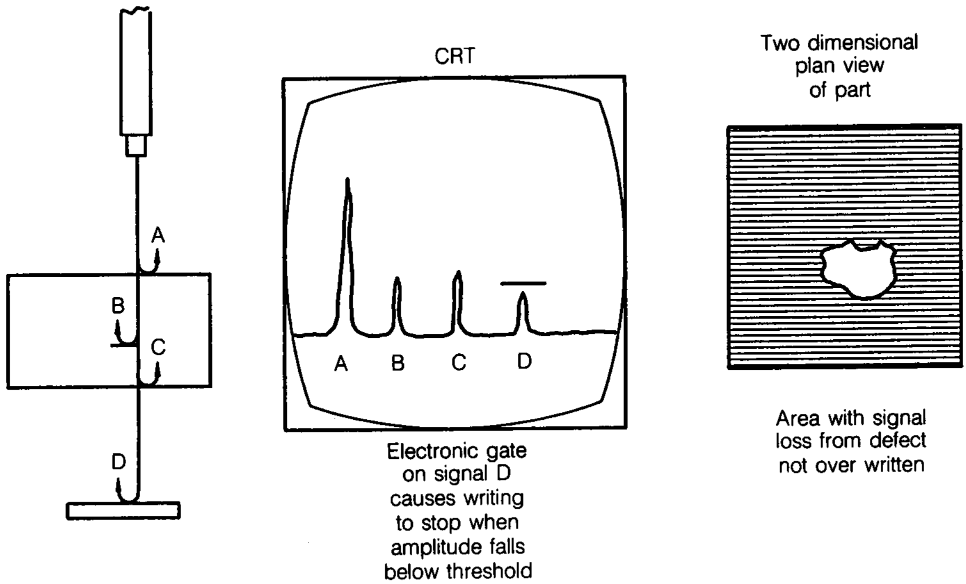


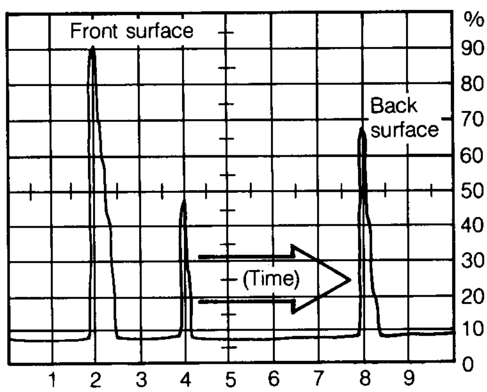
Fig. 9.4.9 Contact Ultrasonic Inspection

(c) C-scan:

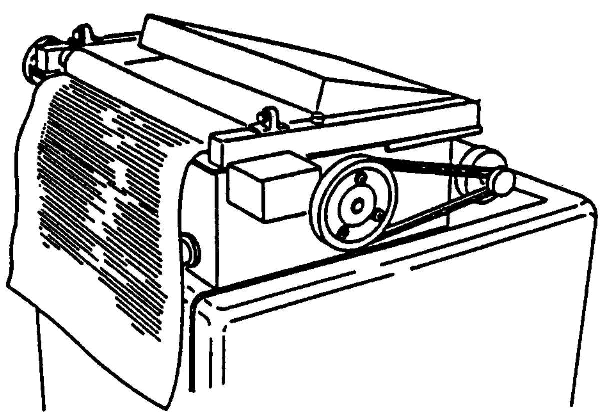
C-scan is a two dimensional plan view recording of data and information taken from the "A-scan" when a part is inspected. All of the information shown on the "C-scan" is first displayed on the scope but is then plotted by a C-scan plotter. There are controls on the scope which allow the technician to determine which signals, and what degree of signal, will be converted into information regarding material quality to be shown on the C-scan's recording [see Fig. 9.4.10(a)(b)(c)]. Fig. 9.4.11 shows a large C-scan machine which uses several transducers, scanning simultaneously, and greatly reduces the time required for C-scan of large parts.



(a) Scope display

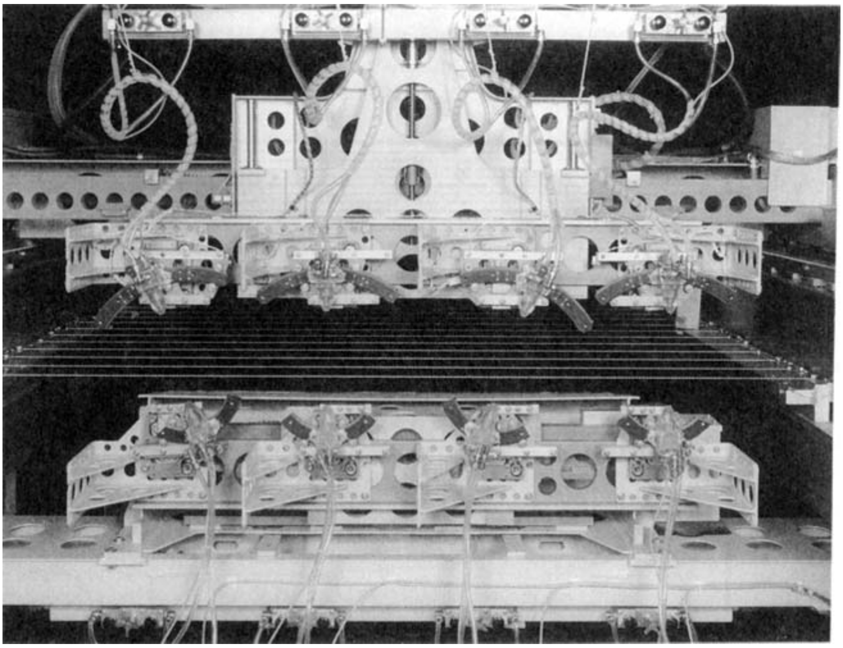


(b) Scope face showing baseline



(c) Plotter (for C-scan only)

Fig. 9.4.10 Ultrasonic Inspection Methods

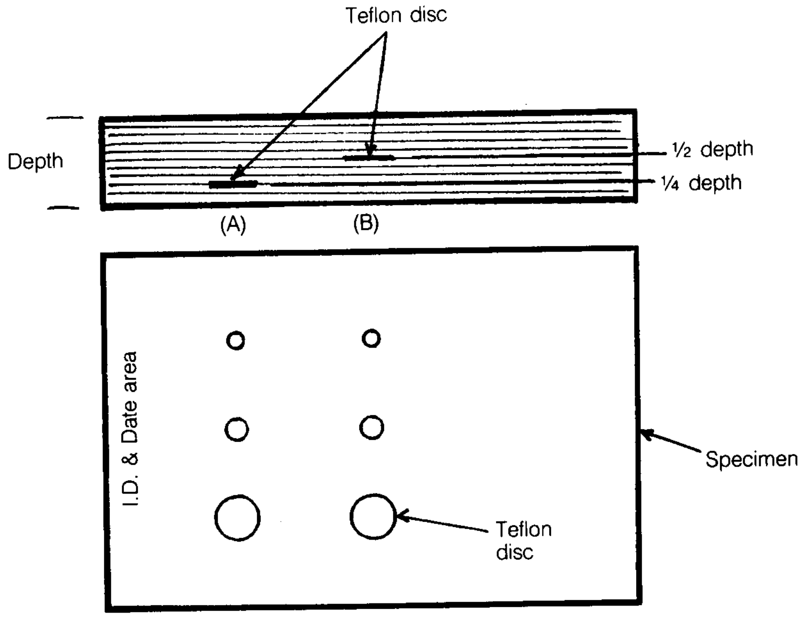


By courtesy of Custom Machine Inc.

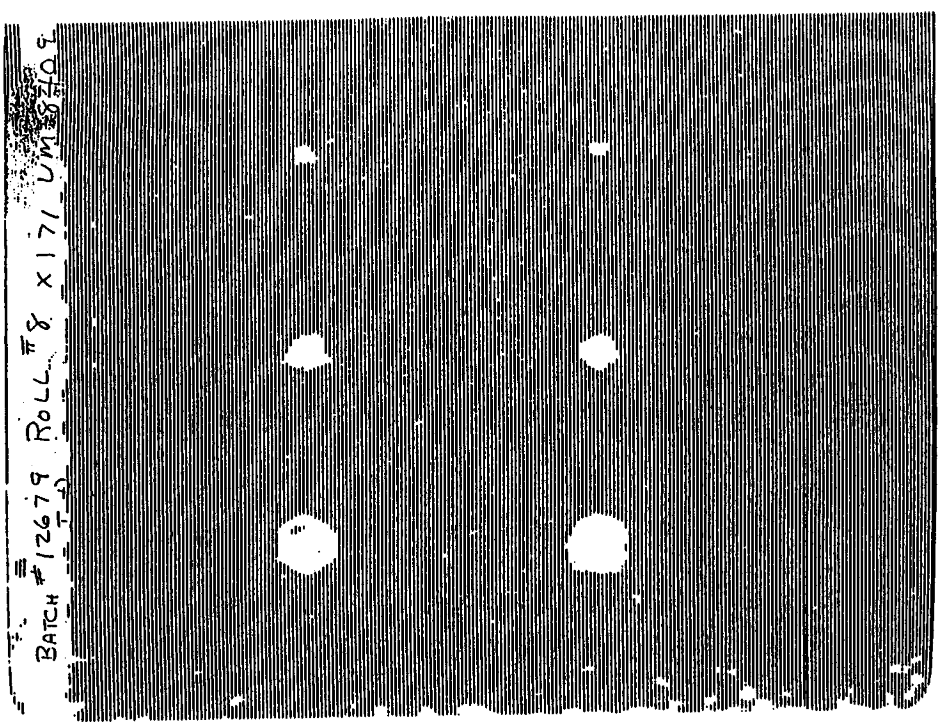
Fig. 9.4.11 C-Scan Of Large Part, Using Several Transducers

It is imperative that a satisfactory reference standard is provided prior to any attempt at performing ultrasonic inspection. This standard must meet the fabrication guidelines and limitations for composite reference standard fabrication. The reference standard is fabricated from the same material, using the same layup method and cure procedures that are used for the test part. The reference laminate standard contains detectable discs (e.g., Grafoil discs with diameters of 0.5, 0.25, 0.125 , etc. inches) which simulate flaws. Honeycomb reference standards usually contain 0.75 or 0.5 inch inserts. The reference standard provides a means for the technician to calibrate the instrument that is used to detect anomalies, and when an anomaly is detected, allows the technician to compare the anomaly to a known flaw (see Fig. 9.4.12). The use of reference standards is summarized below:

- Requirements:
  - Is fabricated from same material and by same process as the part it will be used on
  - Porosity/void content is within specification limits
  - Utilizes Grafoil or Teflon discs in the layup to simulate unbonded areas
- Benefits:
  - Provides a means of sizing defects
  - Provides a means to standardize the calibration of ultrasonic equipment
  - Provides repeatability of calibration



(a) Reference disc imbedded in laminate specimen



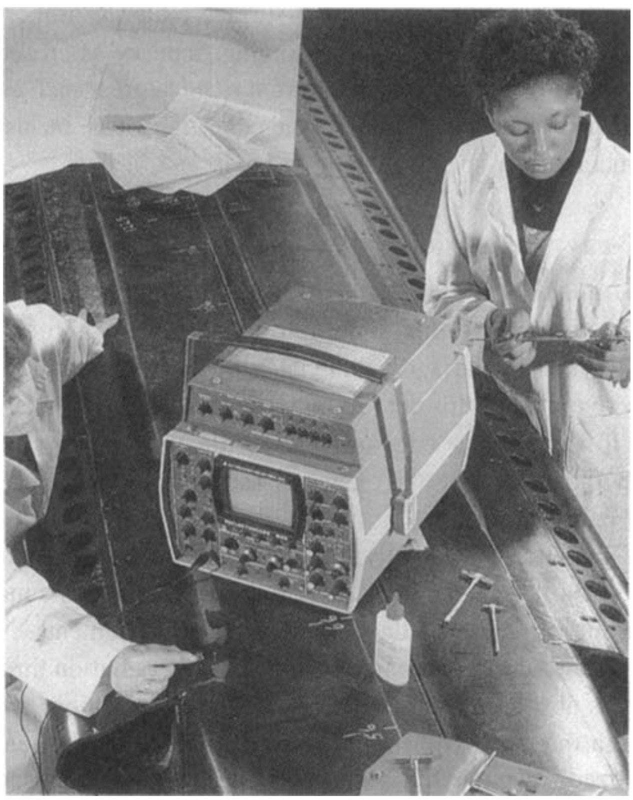
(b) C-scan of laminate specimen

Fig. 9.4.12 Example Of An Ultrasonic Reference Standards

(b) Example of ultrasonic evaluation criteria is given below:
- Single void area — 0.25 in^2. maximum
- Total accumulated area of any detectable voids — 1.0 in^2. in any 1.0 ft^2. area
- Single porous area — 1.0 in^2. maximum
- Total accumulated area of any detectable porosity — 4.0 in. in any 1.0 ft^2. area
- Distance between any detectable defects — 4.0 in. minimum
- Distance of detectable defect from finished edge — 1.0 in. maximum
- Delamination — None allowed

Ultrasonic inspection utilizes sound waves, usually between 500 kHz and 15 mHz. Within certain limits, the higher the frequency, the smaller the detectable flaw. Most inspection are conducted between 2 to 10 mHz.

The selection of an ultrasonic instrument, transducer, and method, etc. is usually followed to inspection specification which should be established in written procedures as to how parts will be scanned (This will help to standardize the inspections being performed). Some equipment is portable (see Fig. 9.4.13); C-scan equipment is not.



By courtesy of Sikorsky Aircraft

Fig. 9.4.13 Portable Ultrasonic A-Scanner Used On Composite Wing Structure

Acoustic Microscopy

The resolution of a C-scan is adequate for most composite structures; however, C-scans are limited by their relatively long wavelength. Acoustic microscopes operate over smaller areas, using frequencies several orders of magnitude higher than C-scan to obtain higher resolution (e.g., 1.0 μm). The higher resolution of acoustic microscopes is often needed when certain parameters, such as fiber population or orientation, fiber/matrix interface bonding [see Fig. 9.4.14(a)], or fiber breakage, are critical.

Acoustic microscopy (also known as Acoustic Micro Imaging — AMI) techniques are basically advanced forms of tradition Ultrasonic Testing techniques and provide greater sensitivity, more precise data gathering and higher resolution acoustic image with faster acquisition times than conventional ultrasonic C-scan systems. Different types of Acoustic Microscopes, SLAM, SAM and C-SAM as described below [see Fig. 9.4.14 (b)]:

(a) SLAM — The SLAM, Scanning Laser Acoustic Microscope, is a real-time, through transmission system and is able to detect defects or material variations throughout the entire thickness of a sample with one acoustic image. Its main advantage is that it can be used to qucikly screen a large number of samples to determine if they are of acceptable or rejectable quality.

(b) SAM — The SAM, Scanning Acoustic Microscope are utilized for the investigation of a sample's surface and near surface material properties.

(c) C-SAM — The C-SAM (C-Mode Scanning Acoustic Microscope) type systems are utilized to investigate a sample for internal features, such as bond or material interfaces, physical defects (cracks, voids, delaminations, inclusions, etc.), and/or microstructure at a specific depth or level.

Radiographic (X-ray) Inspection

Radiography (X-ray) is a useful tool for the inspection of composite parts. Radiography provides an excellent means to examine the interior of honeycomb parts for conditions such as misalignment, missing parts, core damage, inclusions, foam porosity, foreign objects, etc. One effective method for determining ply separations in a laminate is to apply a radio-opaque dye to the laminate edge or damaged area and perform a radiographic inspection. This method can disclose the extent of damage, and to some degree, the plies affected by separation or damage.

The cost of X-ray film is relatively high, but not prohibitive and the method provides the advantage of a permanent record. Portable inspection is suitable for field applications, but is limited because of the critical safety requirements.

The method uses an X-ray source to pass high energy radiation through the test part. Some of the energy is absorbed by the material under examination, while the reminder is transmitted through the part. The energy forms an image on the sensitive X-ray film which, when developed, displays the densities of the material being tested.

Radiographic inspection can be summarized as follows:

(a) X-ray passes high energy radiation through the test part and an image is formed on sensitive X-ray film, displaying the density of tested material

(b) Provides an excellent means to examine interior of honeycomb panels

C-SCAN

C-SAM C-SAM SLAM SAM

Frequency 1 MHz.........10 MHz.........30 MHz.........50.........100.........200.........500.........1 GHz

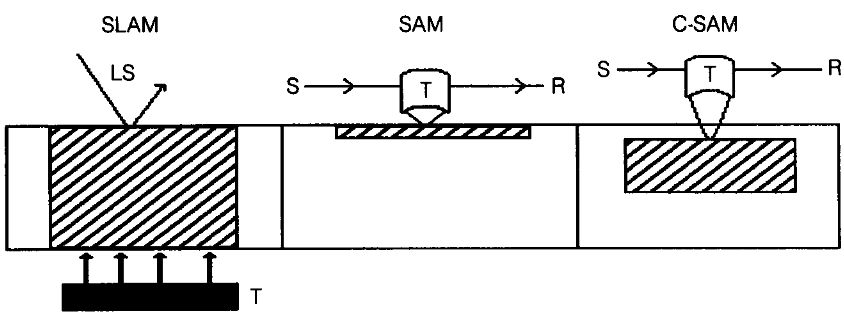
Wavelength 15 mm 15 microns 1.5 microns

—Medical ultrasound—
—Conventional ultrasonic NDT—
—Coarse grain metals: defect detection—
—Cracks in plastic encapsulated IC devices—
—Composite materials—
—Cracks in ceramic IC packages—
—Spot welds—
—Heat sealed food pouches—
—Ceramic chip capacitors: delaminations & cracks—
—Hermetic seal reliability—
—Polymer-foil package delamination—
—Integrated circuit die attach—
—Thick film adhesion and porosity—
—Laser spot welds—
—Fine ceramics: defect detection—
—Fine grain metals: defect detection—
—Seam welds on tin cans—
—Lead bonds on hybrid circuits—
—Ceramic substrate porosity & cracks—
—Cracks in silicon wafers—
—Thin film adhesion—
—Grain structure determination—
—Fine line inspection on silicon—

(Source: SONOSCAN Inc.)

Note: SLAM and C-SAM acoustic microscopes made by SONONSCAN, Inc.

(a) Guide to Acoustic Micro Imaging



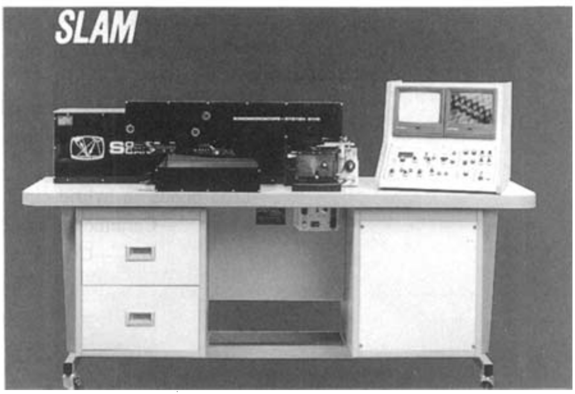
Zone of appliaction within sample
T Transducer
LS Laser Scanner
S Send Pulse
R Receive Pulse

(b) Diagram indicating the various types of Acoustic Microscopy Technologies.

By courtesy of Sonoscan, Inc.

Fig. 9.4.14 Acoustic Microscopy Technology

(c) C-SAM System



(d) SLAM System

By courtesy of SONOSCAN, Inc.

Fig. 9.4.14 Acoustic Microscopy Technology (cont'd)

(c) Laminate ply separations can be detected by applying radio-opaque dye to laminate edge or damaged area and performing X-ray inspection
(d) Provides permanent records on film (see Fig. 9.4.15)
(e) Special precautions must be taken because of potential radiation exposure

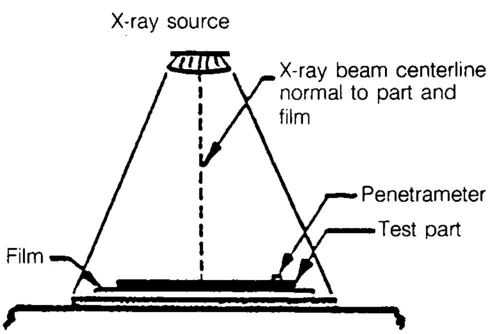
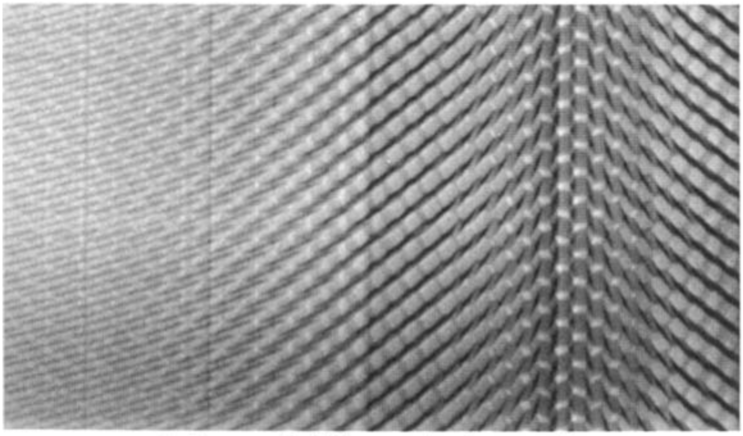
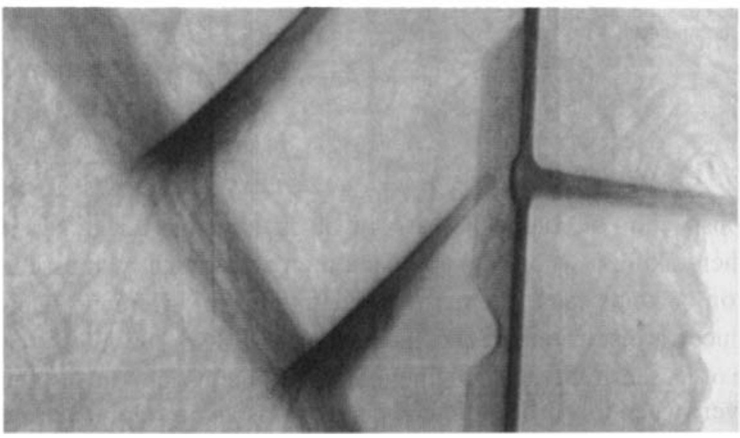


Fig. 9.4.15 Radiography Setup

The real-time radiography systems have been used on composite structure to eliminate slow setup and high film costs; while video-imaging systems are available to enhance the image. Low-energy X-rays improve contrast allowing better inspection of composite honeycomb panels and sandwich structuries, as shown in Fig. 9.4.16.



(a) Structure of a helicopter wing (honeycomb)



(b) Fiberglass material

By courtesy of Lixi Inc.

Fig. 9.4.16 Real-time Low-Energy X-Ray On A Composite Honeycomb Panel

Computer Tomography

Computer tomography is a means of retrieving data which is lost in standard X-ray imaging (which is produced by passing X-rays emitted from a single source only through the object in question). Computer tomography produces a slice of the area of interest, and can produce a 3-D image from a series of slices, as shown in Fig. 9.4.17. Computer tomography takes X-ray images from a number of different angles and processes this information electronically to produce a 3-D image. Theoretically, it cuts the material to examine the interior of the laminate without affecting part functionality.

This method is generally used to examine the matrix laminate, looking for resin-rich or resin-poor regions, or lack of bonding.

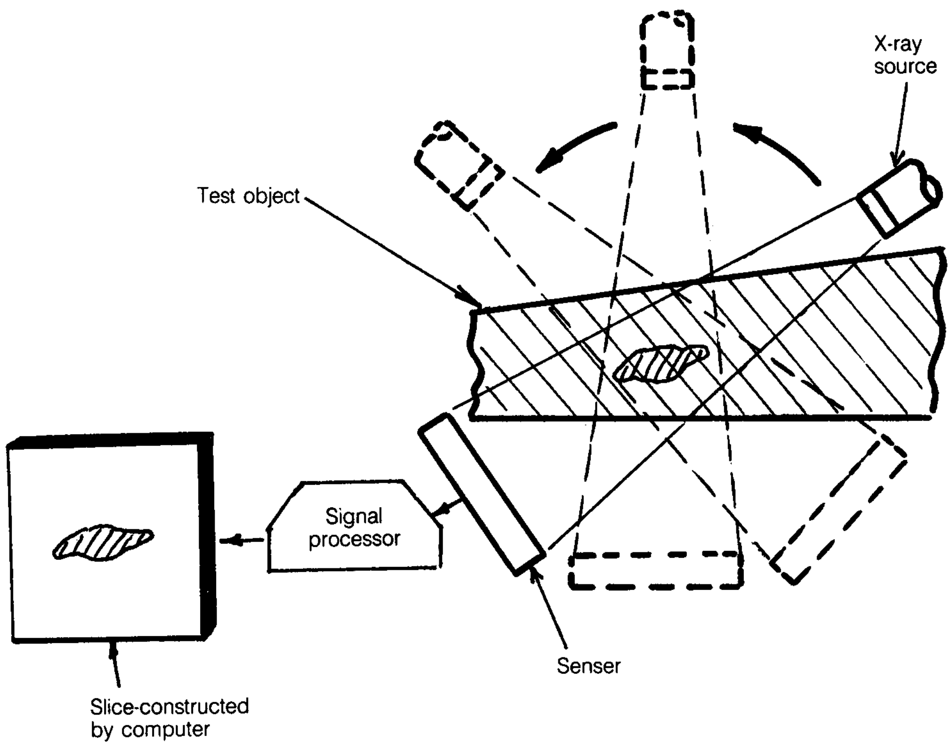


Fig. 9.4.17 Computer Tomography Method

Holography (Laser)

Holographic inspection of composite parts involves taking an image of a part while it is at rest, and then taking a second image when the part is slightly stressed. Defects such as delamination or voids cause the part surface to deform differently than in defect-free regions, produces a distortion, as shown in Fig. 9.4.18, in the interference pattern.

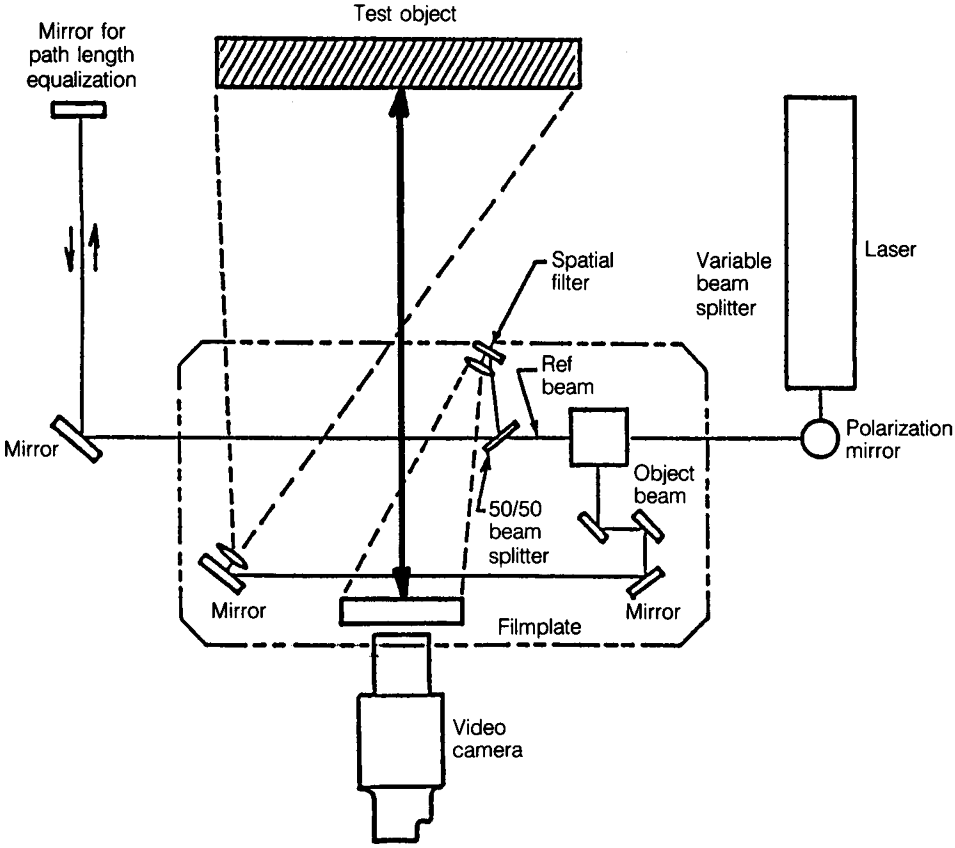
Because of its sensitivity to small movement, problems in fixturing and part stiffness occur. However, when this procedure can be used it is a very rapid and inexpensive way to inspect large structures.

This system allows real-time viewing of holographic images, or recording of permanent images on photographic film for future reference.

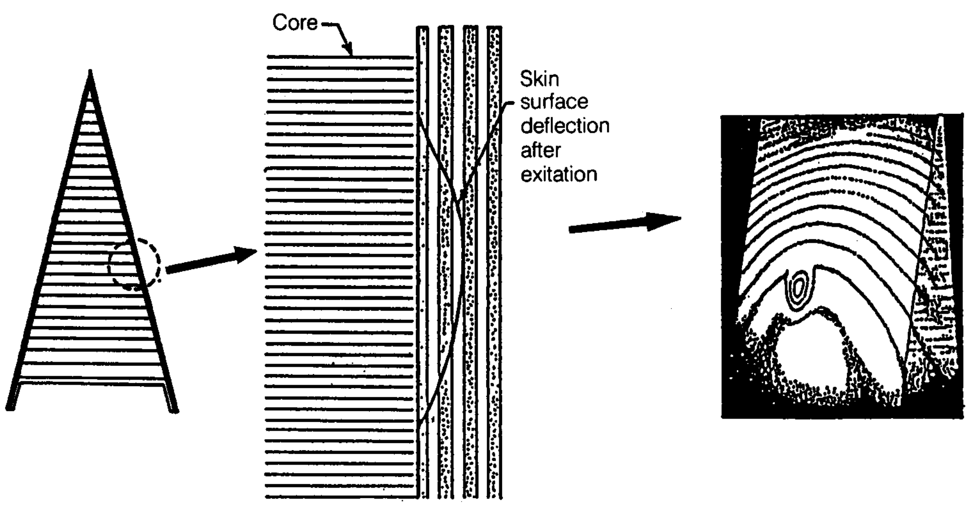
Shearography (Laser)

Shearography (see Fig. 9.4.19) is a laser-based interferometer that is used to detect impact damage, cracks in rivet holes, disbonded or overlapped joints, and to evaluate honeycomb repairs. Moreover, it allows on aircraft inspection without requiring the removal of the structure being tested. Benefits of shearography includes:

- Compared to Holography, no vibration isolation required
- Can be performed in ambient light
- Portablility allows use in field applications
- Real-time imaging
- Non-contact

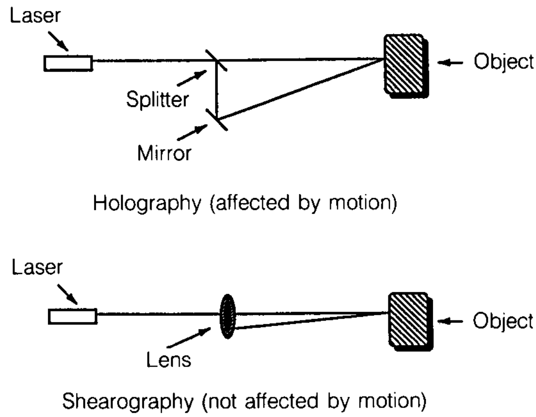


(a) Schematic illustration

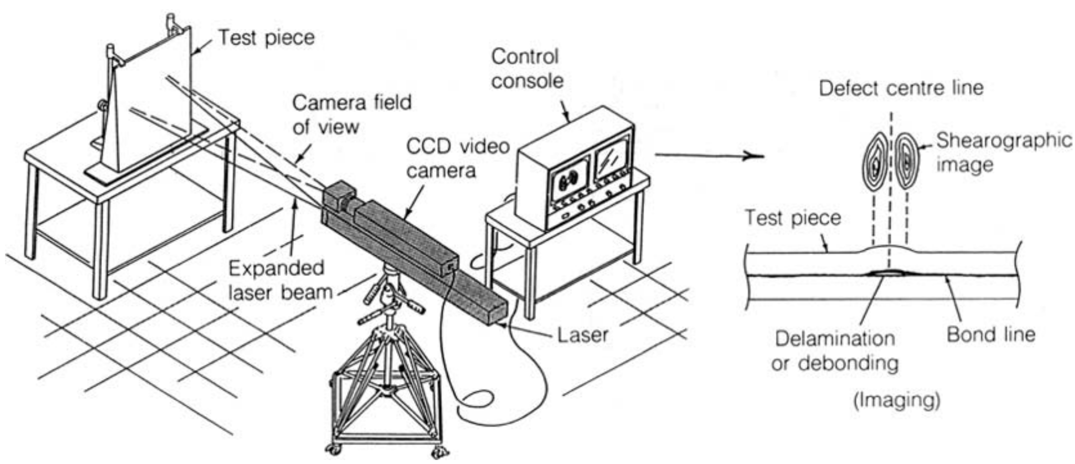


(b) Test specimen (honeycomb wedge)

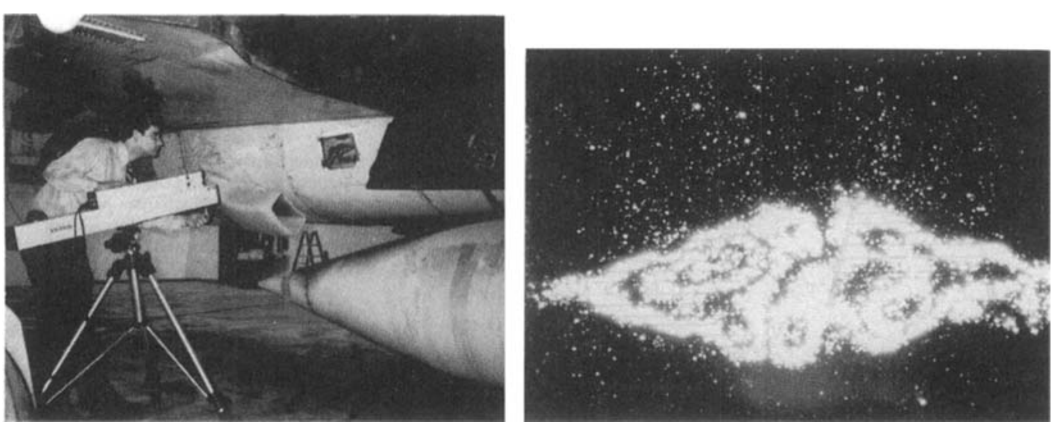
Fig. 9.4.18 Holography Method



(a) Shearography vs. holography



(b) Schematic of shearography arrangement



By courtesy of Laser Technology Inc. (PA)

(c) Shearography equipment to detact area of damage

Fig. 9.4.19 Shearography Inspection Method

Eddy-current

The Eddy-current technique is generally used to detect small surface cracks in metal structures, but this technique can also be used to detect damage in carbon-fiber composites. That is because carbon/fibers are relatively good electrical conductors; the flow of Eddy-current in carbon fibers is impeded where fibers are broken.

Thermography

Thermography is an infrared imaging system which can be used to detect pressurization leaks, thermal-duct leaks, and structural delaminations. It produces high-resolution, real-time thermal images and also has video-recording capability. This system only works on small parts, since providing a heat source for large parts is difficult and expensive.

A summary of NDI capabilities is shown in Fig. 9.4.20.

NDI method \ (Types of defect)	Delaminations	Disbonds	Porosity	Resin variations	Fiber orientation	Core damage	Inclusions	Mislocated details	Fiber defects
Ultrasonics	A	A	A	B/A	C	B	B	C	C
Radiography	C	C	B	B	C	A	A	A	C
Thermography	B	B	B	C	C	C	B	A	C
Dye penetrants	A ⚠1	B ⚠1	B ⚠1	—	—	—	—	—	—

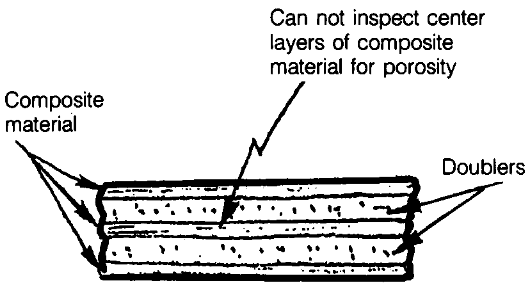
Legend: A — Good capability
B — Fair capaibilty
C — Poor capability
Notes: — No capability
⚠1 — Edges or surface only

Fig. 9.4.20 NDI Capability

9.5 DIFFICULT-TO-INSPECT DESIGNS

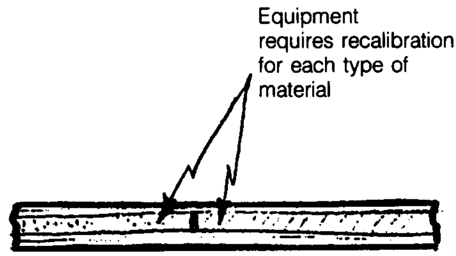
To avoid excessive costs and assure adequate inspection, the composite engineer must be aware of designs which are not readily inspectible by current NDI methods and attempt to avoid them whenever possible. The engineer should coordinate with quality assurance early in the design process to develop adequate inspection processes and standards for designs whioh are not readily inspectible.

Figures 9.5.1 through 9.5.8 show some difficult-to-inspect areas of which the engineer should be aware. This is by no means a complete list, but contains a few of the lessons learned by past composite programs. By having some knowledge of NDI techniques and their capabilities and limitations, the engineer may be able to prevent other difficult-to-inspect parts from being produced.



(It could be inspected dependent on adhesive and doubler material type used)

Fig. 9.5.1 Doublers



(It would be inspected dependent on differences in material types used)

Fig. 9.5.2 Changes In Material Type

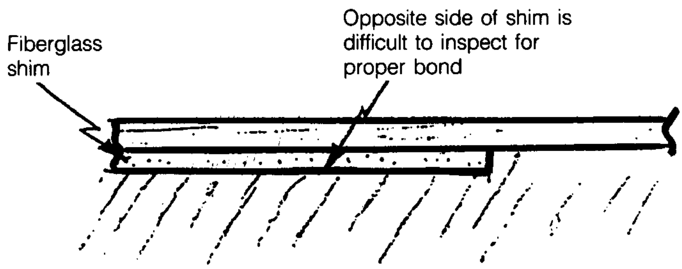


Fig. 9.5.3 Shims

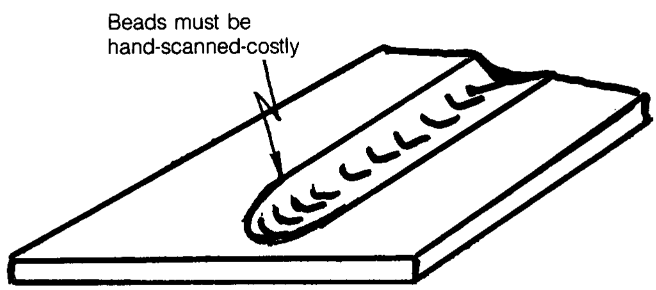


Fig. 9.5.4 Beaded Panels

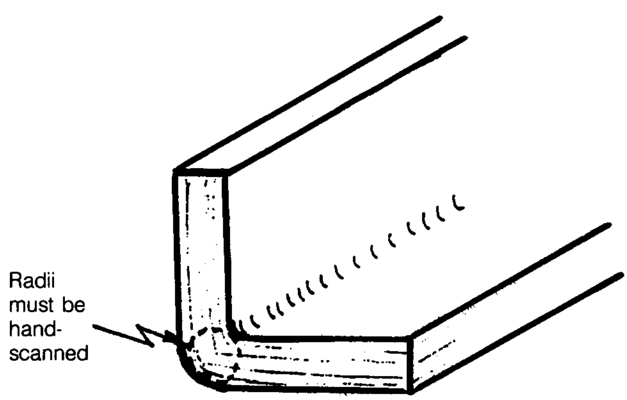


Fig. 9.5.5 Radii

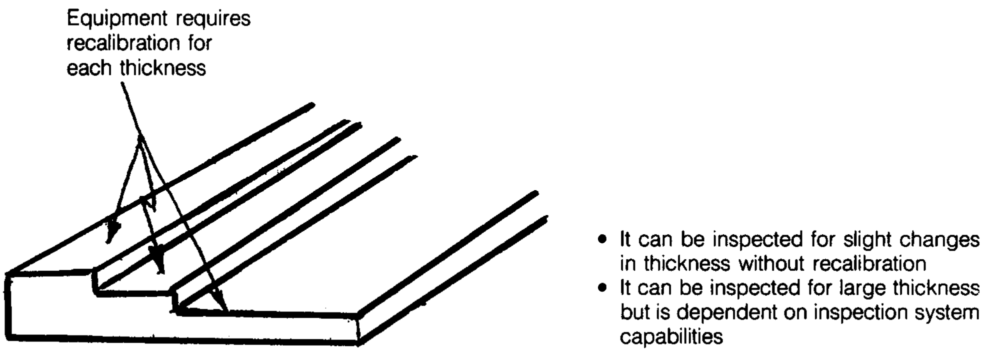
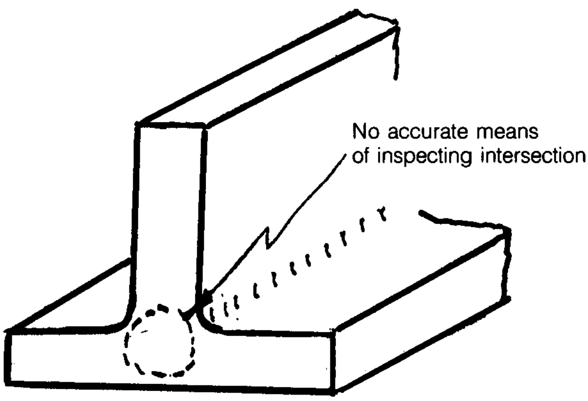


Fig. 9.5.6 Steps/Changes In Part Thichness



(It can be inspected but it is difficult)

Fig. 9.5.7. Tees

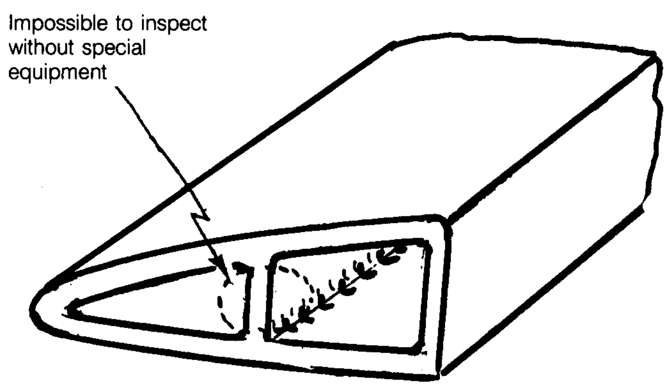


Fig. 9.5.8 Closed Out Areas

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Chapter 10.0

REPAIRS

10.1 INTRODUCTION

The increasing use of composites in both military and commercial aircraft mandates the development of proven repair methods that restore the integrity of the structure. Frequently these repairs must be applied in a timely manner with limited fabrication resources. The basic goal of repair is to restore a part's structural integrity in the shortest time span and at the least cost. The repair should match the strength and stiffness of the original part, while keeping any additional weight to a minimum. Depending on the part, often a trade-off must be made between aerodynamic smoothness and ease of repair. Continued and expanded use of composites critically depends upon the development of repair methods which are both structurally adequate and economically practical. See Fig.10.1.1 for commonly used methods of repair.

This chapter will provide essential information for a basic understanding of composite structures that is not only relevant to repair procedures (see Fig. 10.1.2) but also to the design of repairable composite structures. The design requirements and considerations discussed in Chapter 5 (Joining) are also applicable to repair designs.

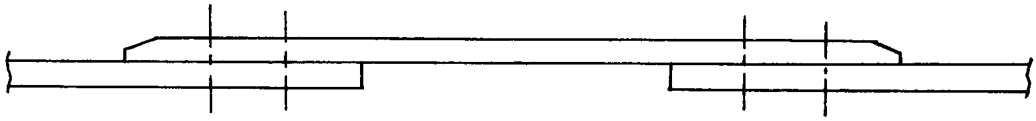
Large part or large area repair often mandates replacement of the part, or its removal for remanufacturing. For damage over smaller areas, repair methods utilize prepregs and wet-layup techniques in conjuction with bonding, bolting and flush patching, etc.
Two major repair techniques (see Fig. 10.1.3) can be categorized as follows:

- Bolted repair
- Bonded repair
  - — Laminate repair
  - — Honeycomb repair
  - — Injection repair of delaminations

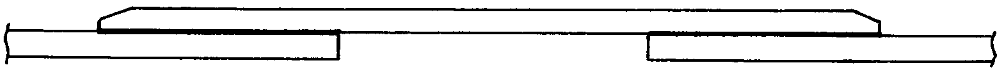
Repair categories are:

- Depot level (military) or maintenance base (commercial) —Performed at major maintenance bases or the manufacturer's facility
- Field level (military) or line station (commercial) — Performed at a forward operating base with limited facilities

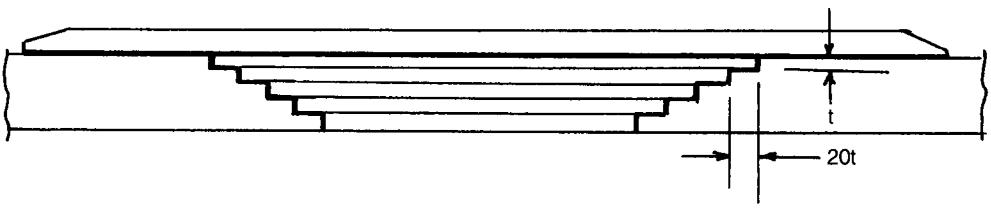
Even with depot repair it may still be desirable to make the repair right on the aircraft to reduce downtime. This also lessens the chance of incurring further damage by removing the part.



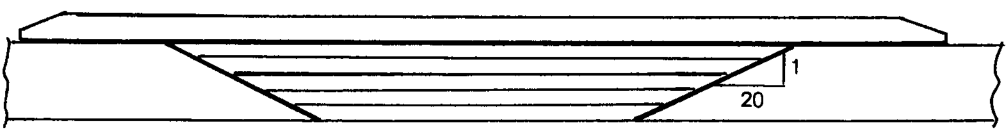
(a) Bolted repair



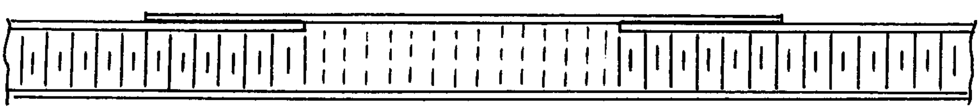
(b) Bonded repair



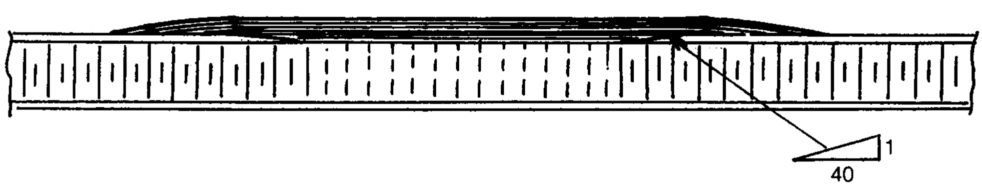
(c) Stepped-lap repair



(d) Scarf repair



(e) Honeycomb repair



(f) Alternative honeycomb repair

Fig. 10.1.1 The Most Common Methods Of Composite Repair

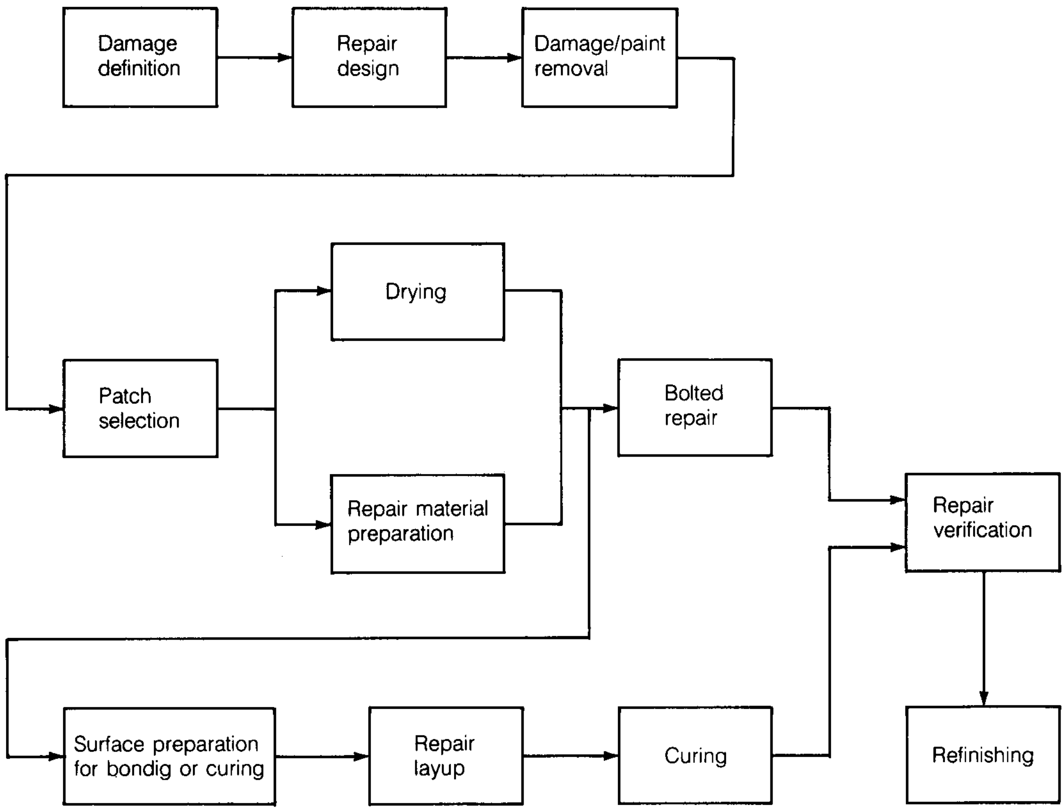


Fig. 10.1.2 Repair Process

Repair method	Pros	Cons
Adhesive repair	• Lighter weight • Distribution of load over wider area • Better for thin laminates	• Degradation in service due to temperature and humidity • Requires surface preparation • Difficult to inspect • More difficult to perform correctly
Mechanically fastened repair	• Not adversely affected by temperature and humidity • Less surface preparation required • Easy to repair and suitable for field repair • Generally visual inspection required, and sometimes may need Eddy-current to inspect holes	• Adds weight and bulk • Requires holes in weakened members • Produces stress concentration • Clearance between bolt and hole distributes load unevenly • Susceptible to delamination damage from driling/machining

Fig. 10.1.3 Bonded Vs. Bolted Repairs

Regardless of the care with which any structure is utilized, the possibility of its requiring field repairs must be borne in mind during the design phase. The ability to control the pressure and temperature of complex cure cycles is severely limited when attempting a local repair in the field. This may well force the repair procedure to be quite different from the original fabrication. Ensuring that repairs of adequate structural integrity can be made must be considered throughout the design process. Only rarely will the repaired part have the structural integrity of the original; in fact, field repairs which results in less-than-original strength are considered satisfactory for some military airframe structures. It is frequently difficult to apply greater than atmospheric pressure during repair curing, unless an autoclave is available, and this clearly influence repair quality.

Bonded repairs have the advantage of not requiring additional cut-outs in the damaged component, and the only surface preparation required for the repaired part is solvent wipe, drying, sanding, and cleaning out of damage. The clean-out may only involve applying a room temperature curing resin to hold down loose fibers. An external cure-in-place patch can be readily applied, and with vacuum cure this is a procedure easily adapted to most field level facilities. This is the repair approach which can be most easily accomplished on a contoured surface. There is a significant reduction in the mechanical properties of prepreg with vacuum cure (Depends on cure temperature as well), however, and an autoclave cure whenever possible, will provide superior properties.

When aerodynamically smooth repairs are required, a cure-in-place repair is used with a step-lap [see Fig. 10.1.1(c)] or scarf repair [see Fig. 10.1.1(d)]. This is a structurally efficient, but expensive and time-consuming repair process. If back-side access exists (not a common case) and there is no interference, an external patch can , of course, be applied to the inner surface thus preserving aerodynamic smoothness.

A problem encountered in many field repair situations is the lack of storage facilities for frozen composite materials. Use of C-staged prepreg, which can be stored at room temperature but still has flow characteristics necessary for proper cure, is one approach which looks promising. The use of two-part wet layup resins, analogous to current fiberglass repairs, would eliminate storage problems; but these systems are generally have high porosity with reduced durability compared to structural prepreg systems.

The effective composite repair requires knowledge of the following:

- Application and control of heat/pressure/vacuum systems
- Material behavior related to coefficient of thermal expansion
- The different characteristics exhibited by metals and composites
- Classes/types of adhesives/resins and their intended service environment
- Classes/types of fiber reinforcements and their use in prepregs and wet layup
- Characteristics of honeycomb core materials.
- Surface preparation techniques
- The creation of test specimens which are used to verify the strength of the repaired structure (see Fig. 10.1.4)

Criteria

The repair must restore the capability of the part to withstand design ultimate loads (without limitations unless otherwise specified) and must restore the full service life of the part. Design strength is based on the notched strength of composite in order to account for fastener holes and undetected damage. As part of the certification of a new aircraft, the FAA and military specifications require a satisfactory plan for accomplishing airframe repairs.

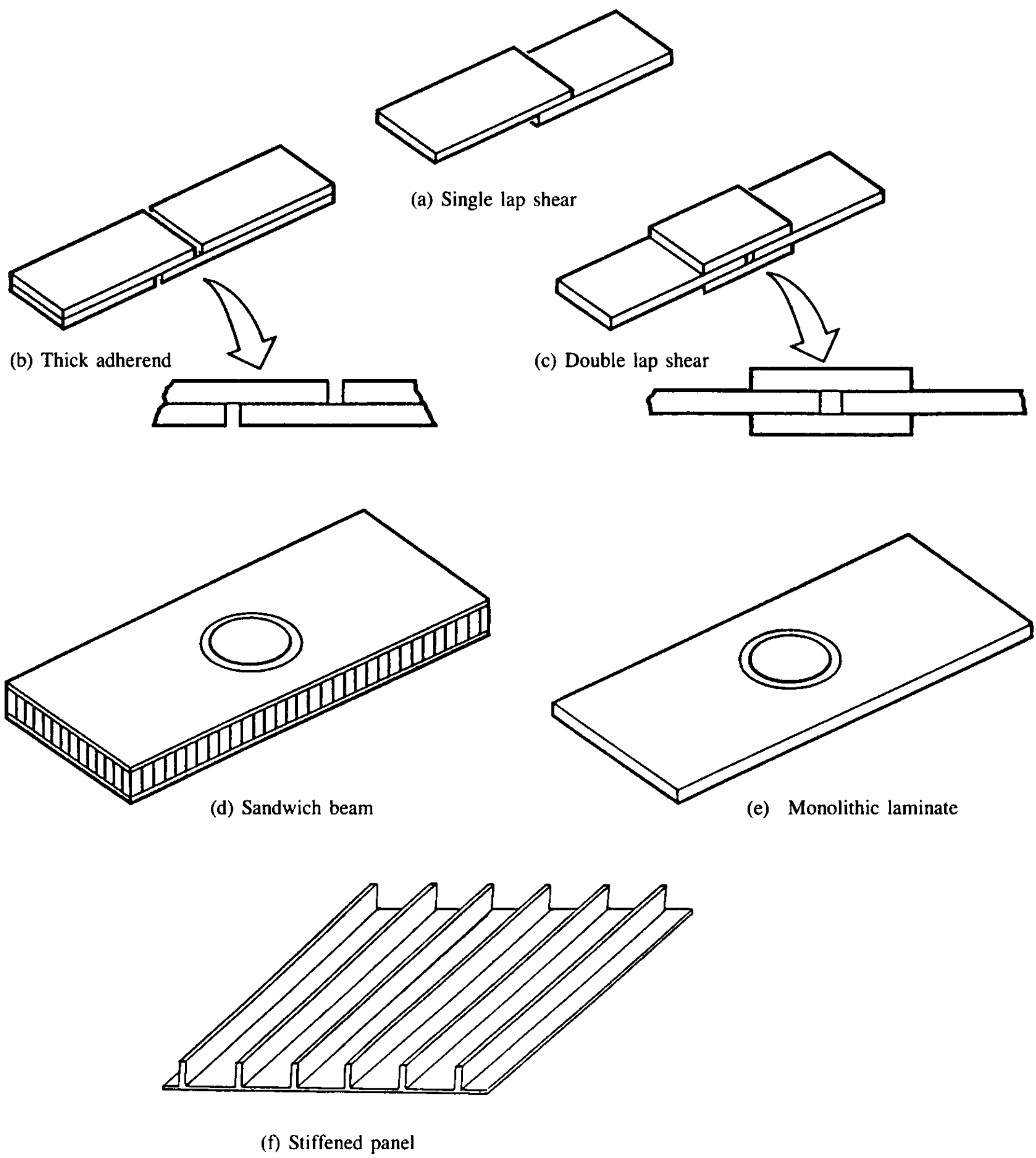


Fig. 10.1.4 Repair Test Specimen Configurations (For Both Bolted and Bonded Repairs)

(1) Common criteria for repairs include:
- Static strength and stability:
  - Full versus partial strength restoration
  - Stability requirement
- Durability for the life of the part:
  - Fatigue loading
  - Corrosion
  - Environmental effects
- Stiffness requirements:
  - Deflection
  - Flutter
  - Load path changes
- Aerodynamic smoothness:
  - Manufacturing criteria
  - Performance degradation
- Weight and balance:
  - Effect of size of repair
  - Mass balance effect on dynamic response
- Operational temperature:
  - Low/high temperature requirements
  - Effect of temperature on materials
- Environment:
  - Types of exposure
  - Moisture effects on repair materials
- Related airframe systems:
  - Fuel system sealing
  - Lightning protection system continuity
- Cost and schedule:
  - Downtime
  - Facilities and equipment
  - Skill personnel
  - Material handling

(2) Aircraft battle damage repair (or field repair):
This repair is a quick fix that has little in common with the type of permanent repairs usually made to both commercial or military airframe structures. Rather than restoring an airframe to meet the original condition, this repair focuses on getting a battle aircraft back in the air for one or two more flight. The repair might be good enough for several flights, but is not intended to last for years. Battle damage repair does not use the same materials, repair designs or level of complexity as peacetime repairs.

Aircraft battle damage repair does not have to restore the structure to its exact factory specifications; however, repaired surfaces are made as smooth as is possible. The aircraft may use more fuel than normal but at least it can be flown. A battle damage repair procedure or manual is used to control the repair procedures.

Classification of Damage

Most repairs relate to three types of damages:

- Manufacturing anomalies — Voids delaminations, surface depression, etc. during curing process
- Mishandling damage — Aircraft and parts are damaged on the ground (impact damage resulting from ground handling is the primary cause of damage to composite structures, see Fig. 10.1.4)
- Environmental damage — Hail, lightning strikes, bird strikes, debris (kicked up on takeoff or landing), etc.

Damage is defined as any deformation or reduction in cross section of a structural member or skin of the aircraft. Damage can consist of a hole, wrinkle, twist, nick, crack, scratch, delamination, or areas which are corroded (usually galvanic corrosion between metal and carbon composites) or burned. In general, damage to the structure is classified as either negligible, repairable, or necessitating replacement according to the following definitions:

(1) Negligible damage:

Damage to the airframe which does not affect the strength, rigidity, or function of the part involved is classified as negligible. Negligible damage is defined as the removal or distortion of material which can be permitted to exist as it is or which may be corrected by a simple procedure, such as the smoothing out of nicks or scratches.

(2) Repairable damage:

Repairable damage is divided into two basic categories as follows:

(a) Damage repairable by patching — Damage which can be repaired by installing a patch across the damaged portion of a part is defined as damage repairable by patching. The patch is attached to the undamaged portions (on all sides of the damaged portion) of the part to restore the full load-carrying characteristics and airworthiness of the airframe. Damage repairable by patching is specified, if practicable, for each member of an airframe on the material identification and repair index illustrations included in the structural repair manual.

(b) Damage repairable by insertions — Damage which has to be repaired by cutting entirely through a part and inserting a length of member identical in shape and material to the damaged part is defined as damage repairable by insertion. The member spliced into the space resulting from the removal of the damaged portion is identified as the insertion member. Splice connections or patches at each end of the insertion member provide load-transfer continuity between the existing part and the insertion member.

Generally, insertion repairs are used to simplify repair when the damaged portion is relatively long, when the damaged member has a complex shape, or when interference between the repair members and adjacent structural member is to be avoided.

Surface impact	
Dropped tools	Relatively common occurrence, although seldom specifically documented. Majority of hand tools weigh less than one pound although power tools of related equipment may be considerably heavier. Height of drop depends on height of work stands which may be used adjacent to the part impacted.
Dropped equipment	Although not specifically reported, dropping of mechanical, electrical or hydraulic equipment or access doors during installation or removal is a potential cause of impace damage. Weight and height as well as degree of risk depends on the specific aircraft.
Maintenance stands	Cases are reported in which a corner of a maintenance stand is pushed or falls against an aircraft component. Typical damage is a puncture or a surface gouge, although severity can vary widely.
Major damage	Several cases are reported in which a vehicle (truck, forklift, etc.) drove or backed into an aircraft. Since no reasonable degree of protection can be effective against this, ease or repair or replacement of susceptible components is necessary.
Edge and corner impact	
Dropped part	Frequently occurs for parts which are removeable from aircraft especially if the part is heavy or awkward to handle. Corners most easily damaged when striking flat surface such as concrete floor. Edges may be damaged but less frequently when striking other object, e.g., work bench corner, door frames, etc. Impact energy depends on wieght of component, height of drop, velocity at impact, incidence angle and resistance of object struck.
On-aircraft impacts	Impacts are reported due to work stands, hoisting cables or other equipment striking the exposed edges of fixed panels or opened doors. Appears to occur less frequently than dropping of removeable parts. Severity can vary widely from negligible to severe.
Walking/Heel damage	
Local pressure	Numerous cases are reported of dents believed to be caused by walking. Must frequent near intended walkways. Can result from heel presure or other foregin objects.
Walking	Disbonding of face sheets from honeycomb core has been observed and may be caused by walking, although this casue has not been definitely established.
Fastener hole wear	
Fastener shank abrasion	Retaining ring grooves in the shank of quick release fasteners have caused hole elongation. Condition is aggravated by a lateral force on the fastener during installation or removal, Force tends to be caused by dead weight of component, misalignment of fastener, etc. No hole elongation noted with smooth shank fasteners.
Pull through	Wear under countersunk head and local cracking and delamination can result if gasket or substructure configruation permits high tensile force in fastener.

Fig. 10.1.5 Causes of Damage (Ground)

(3) Damage necessitating replacement:
Damage which cannot be repaired is classified as damage necessitating replacement. Damage to parts which are of relatively short length, unless defined as negligible, should be considered as damage necessitating replacement because the repair of short members, generally, is impracticable. Some highly stressed members cannot be repaired because the repaired member would not have an adequate margin of safety. Such members, when damaged beyond negligible limits, must be replaced. Members with configurations not adaptable to practicable repair procedures must also be replaced when damaged.

10.2 BOLTED REPAIRS

The bolted repair method is not new and it is borrow from conventional metal sheet metal repair. Probably the quickest repair method is to bolt a patch over the damaged area. Plates of metal, such as aluminum (not to carbon composite skin) or titanium, or precured composite laminates can be bolted into place with little or no surface treatment.

Bolted repairs eliminate many of the facility problems and limitations of bonded repairs, but do require additional cut-outs which is structurally undesirable. The drilling operation, particularly where back-up cannot be provided, can result in additional damage or an oversize hole which will not pick up load properly. Special drills (see Fig. 10.2.1) are available, however, which minimize this problem. Bolted repairs are better adapted to field level facilities than bonded repairs, but field repairs must typically be accomplished without back-side access, requiring the use of blind fasteners. Improved blind fasteners have been developed for composites, and are currently being used. Fig. 10.2.2 shows both aluminum and titanium patches typically used to repair a 4.0 inch (10.16 cm) diameter damaged hole.

Repair patch material selection considerations:

- Titanium — Titanium sheet (6Al-4V) is a good patch material because of its corrosion resistance and high stiffness-to-weight ratio
- Aluminum — Aluminum sheet is generally more readily available under field repair conditions and is more easily machined than titanium, has a lower density, and can be protected against corrosion
- Stainless steel — Stainless sheet has corrosion resistance and high stiffness, but has a high density and is difficult to machine; it is generally not used
- Precured carbon/epoxy laminate — This laminates may be feasible for both bolted and bonded patches
- Woven fabric for wet layup process
  - Use 5 harness 12 × 12 in^2, weave (6,000 fibers/tow)
  - Impregnate on-site with resin or adhesive

The thickness requirements for bolted repair should take the following into consideration:

- Minimum thickness — No feather edges (knife edge) should be allowed if countersunk fasteners are used

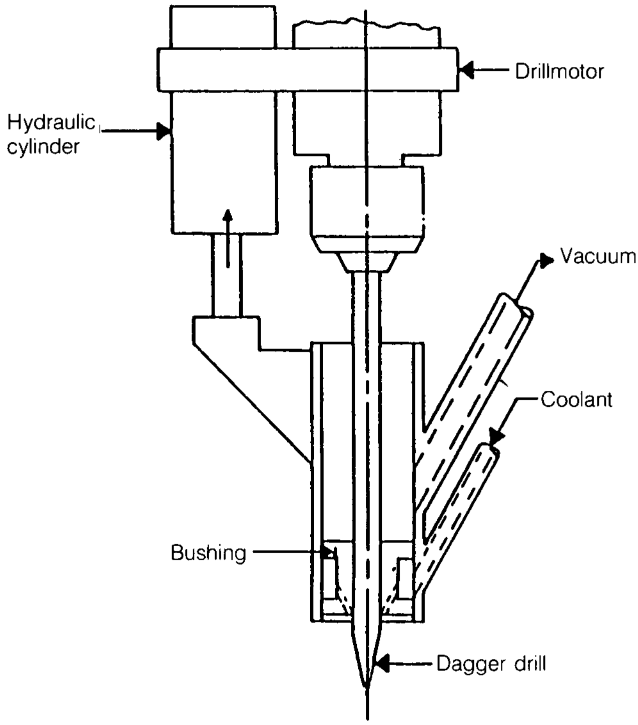
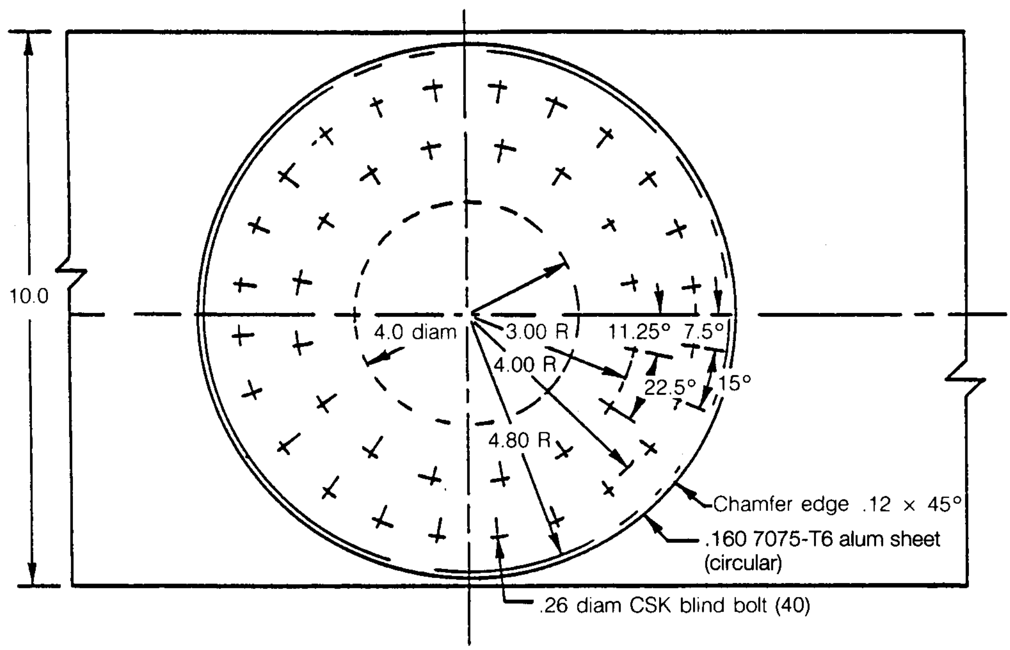
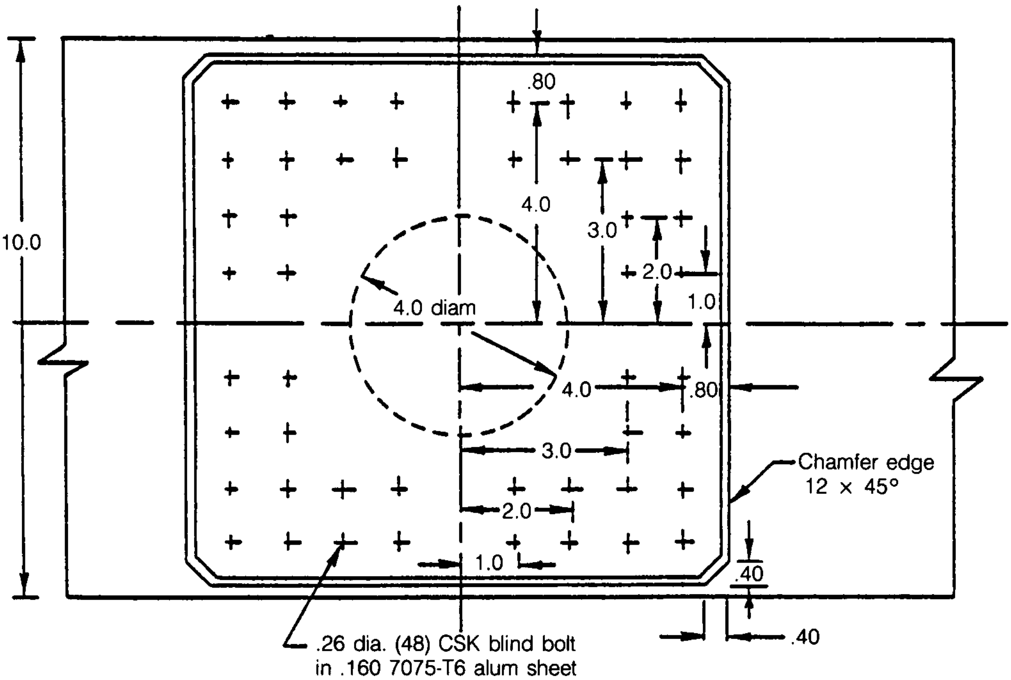


Fig. 10.2.1 Drill And Countersink System Schematic



(a) Circular patch

Fig. 10.2.2 Aluminum Patches for Bolted Repair (Parent Laminate — 24 Plies Graphite/Epoxy)



(b) Rectangular patch

(Dimensions are in inches)

Fig. 10.2.2 Aluminum Patches For Bolted Repair (Parent Laminate — 24 Plies Graphite/Epoxy) (cont'd)

- Maximum thickness — Because of the requirement for surface aerodynamic smoothness, a maximum patch thickness [such as 0.16 inch (4.06 mm)] should not be exceeded in bolted repair (also bonded repair)
- Compexity of analysis with stepped straps or doublers; lots of field fasteners

There are drawback because this repair is not tailored to the needs of composite structures. Metal patches are usually heavier than the composite materials they are replacing, and they can alter the radar cross section.

Bolted repairs are simple because minimal equipment is needed to do the job and neither refrigeration nor heating blankets are required. Bolted repairs are the most common for battle-damage repairs because of reliability and relatively fast application to a damaged part. The design of bolted repair must consider the following:

- Patch material, thickness, and shape
- Fastener type, material, head configuration, and shank diameter
- Geometric arrangement of the fastener pattern
- In military aplications, no back-side access is assumed, so blind fasteners must be used (see Section 5.2 in Chapter 5)
- Recommended drilling procedures:
  - Wet installation of fasteners and use of faying surface sealant in carbon/aluminum interfaces is mandatory

  - Aluminum patch — Drill undersize pilot holes in aluminum, locate on composite with clecos or clamps, and use patch as template to drill full-size holes in composite
  - Titanium patch — Drill full size holes in titanium, use as template to drill composite
  - Drilling operations must be separate or used special drill (see Fig. 5.2.38 in Chapter 5), if metal patch is used
- Where possible it is best to use existing bolt holes
- Bolt holes weaken structures
- Bolts in a blind hole can pull out
- For critical or highly loaded structures — fastener locations must be precisely defined by structural analysis to achieve recovery of design strength
- Effects of fastener clearances must be considered
- Blind fasteners may have limited effectiveness for repair because of reduced pull-off strength
- Fasteners must be wet installed for aluminum (also recommended for titanium) patches because of galvanic corrosion and moisture or water intrusion problems

The following rules are recommended for patch bolt patterns (see Fig. 10.2.2):

- Each bolt is spaced no closer than approximately four bolt diameters to an adjacent bolt three diameters to any edge of the laminate
- The edge distance for metallic patches is should keep two diameters
- The damage must be surrounded by at least two rows of bolts to protect against biaxial or off-axis load and inadvertent oversize bolt holes

10.3 BONDED REPAIRS

Bonding a patch usually is more reliable than bolted repair because bonding produces no holes, which are regions of increased stress. Both metal and composite patches can be bonded over a damaged area. External patches are made of either aluminum, titanium, precured laminate, wet, or prepreg layup. For metal patches, titanium is commonly used because of its non-galvanic reaction with the parent carbon/epoxy skin and its high strength and stiffness to weight ratio. The patch functions as a seal over the underlying repair and structurally carries some of the applied loading which the repaired part or component is expected to experience. Bonding methods require more rigorous control of surface treatments, storage of heat-sensitive bonding materials, and special equipment and is expensive.

Field repair environments generally restrict users to vacuum-bag cure techniques, and localized heat sources such as heat-blankets or pads, infrared lamps, or hot air guns, are generally used. Although epoxies are preferred by many for repair systems, their inherent reactive nature and potential storage problems make thermoplastics attractive as bonding agents. However, the disadvantage with thermoplastic is that they all require processing temperatures in excess of 500°F (260°C) and can not be used on themoset structures (250°F service temperature).

The design of a bonded repair must take the following into consideration:

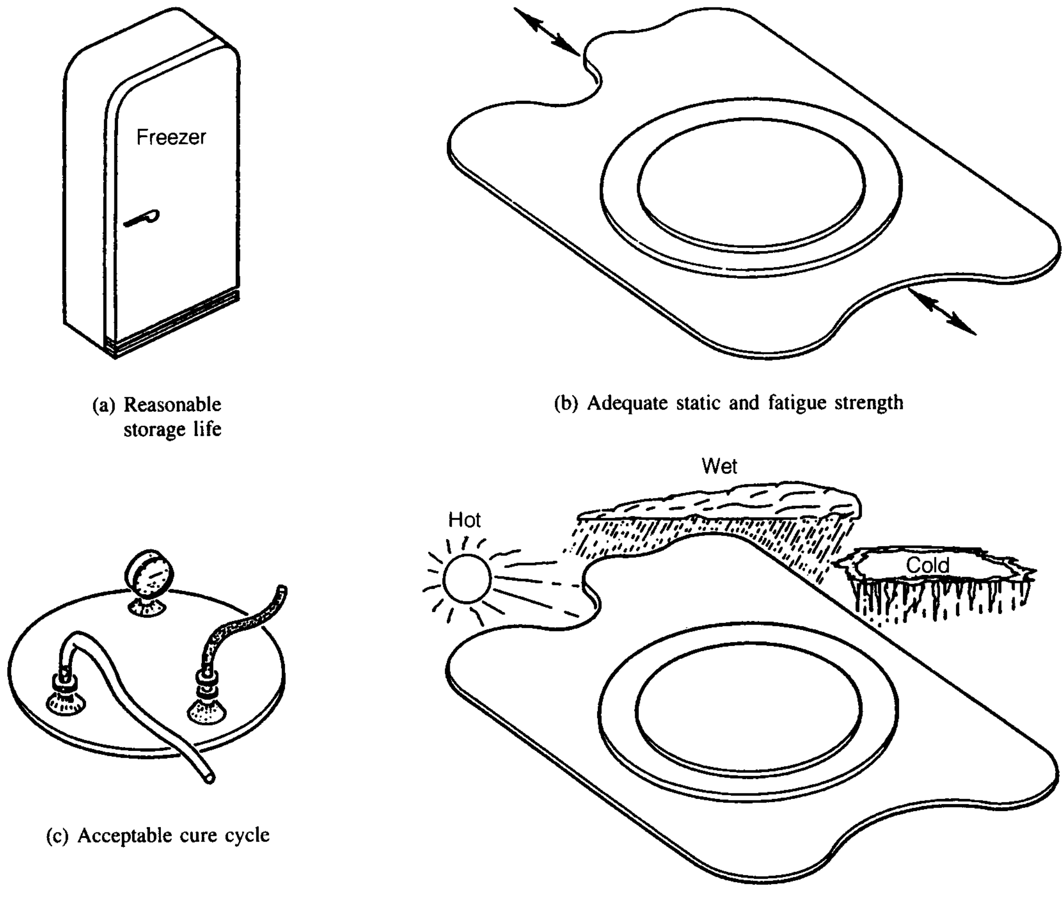
- Patch material, thickness, shape and step taper
- Surface preparation
- Adhesive or resin requirements (see Fig. 10.3.1)
- Room temperature storage ability (because of the logistical problems of resupply for field repair)
- Control of cure cycle
- Storage history of adhesive and treated patch materials
- Specialized skills and equipment required
- Vacuum pressure (see Fig. 10.3.2) adequate for bonded and cure- in-place repairs
- Cure-in-place repair (see Fig. 10.3.3) is advantageous for curved surfaces
- Structural advantages of bonded repair:
  - — No cutout required
  - — Can restore higher strength levels than bolted repair in thin laminate skin
- Flush aerodynamic bonded and cure-in-place repairs provide the most effective strength recovery; can restore design strength for thick high load laminates
- External bonded and cure-in-place repairs are adequate for restoration of design strength in lightly loaded structures
- Boron is more difficult to drill or cut then carbon, so bonded repairs are preferrable to bolted repairs

The alternative repair considered for the small patch include metal foils [titanium or aluminum (not used on carbon composite skin) with thickness of 0.008 inch (0.203 mm)], and several precured forms of composite sheets. Using a stack of progressively smaller foil sheets minimizes the peak adhesive shearing stresses and minimizes peeling stresses at the edge of the patch. Treatment of titanium or aluminum to provide a bondable surface is not feasible in shipboard environments. Etched and primed foils must be supplied in kit form for bonded repairs.

Fig. 10.3.3 shows a titanium patch which consists of several layers of precured composite plies and titanium foil, e.g., 0.008 inch (0.2 mm) thick or 0.016 inch (0.4 mm) thick,that are bonded to each other, and then to the damaged skin.

Care must be taken not to overheat the laminate being repaired. Overheating causes moisture to expand, forcing the plies to delaminate. Too much heat in one place can cause skin blisters. Constant temperature heat blankets provide a reliable cure, and virtually eliminate overheating problems when used correctly by trained operators. Heat is typically applied by means of flexible silicone heating pads laid directly on to the repair (see Fig. 10.3.4). These pads have sophisticated thermal controls for precise staging of the cure cycle. The time/temperature/pressure cure cycle can be monitored manually or automatically with a programmable repair console as shown in Fig. 10.3.5.

Fig. 10.3.6 shows a heating system (Moen Systems) which can deliver hot air to the repair area at temperatures up to 1500°F (816°C). This system expels high-velocity, low-pressure air to the repaired surface and also effectively dries parts.



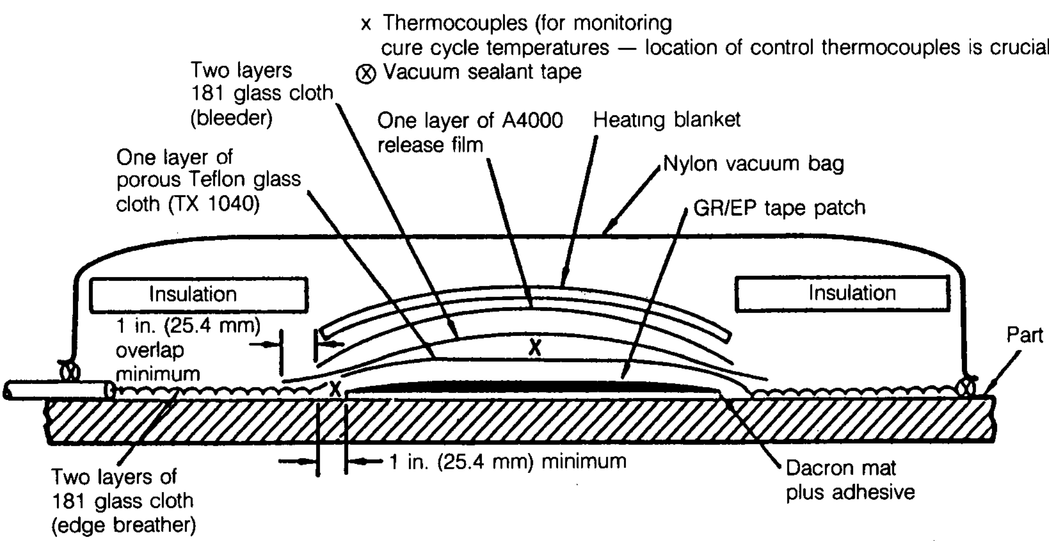
(a) Reasonable storage life

(b) Adequate static and fatigue strength

(c) Acceptable cure cycle

(d) Environmental resistance

Fig. 10.3.1 Adhesive and Resin Requirements

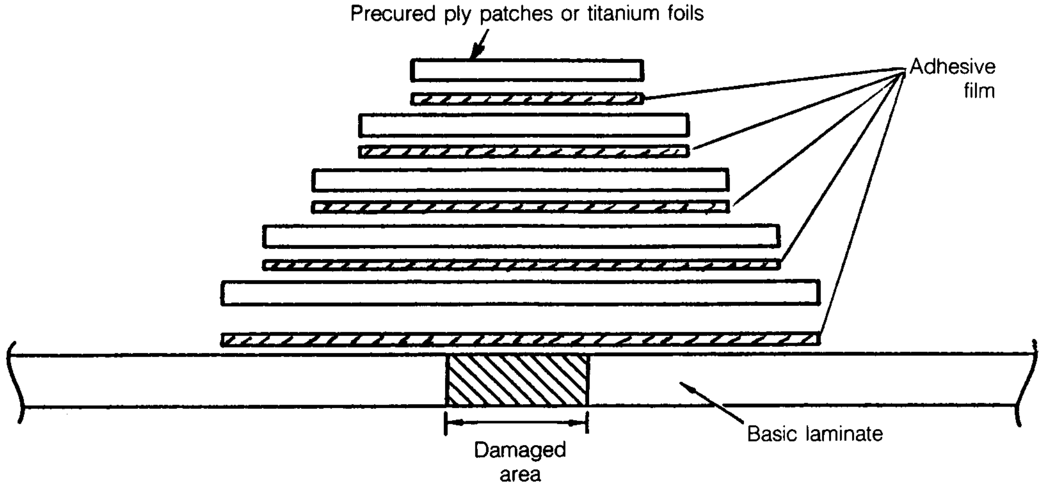


(Note: Insulation is preferably placed and arranged over the vacuum bag such that the patch temperatures are within the specified cure temperature range)

Fig. 10.3.2 Typical Repair Bagging Arrangement

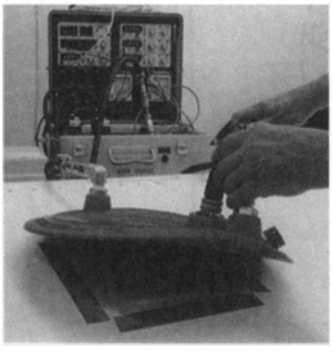
Bonded repair can be either cure-in-place or precured plies used with an adhesive. Sometimes metal and/or precured and cocured plies are combined in the same repair laminate:

(a) Cure-in-place — The entire patch is cocured to the part in one operation (see Fig. 10.3.7) or in thicker laminate or on vertical surfaces multiple cures may be performed; the part may not be able to withstand both the pressure and temperature of an autoclave, so lower pressure and temperature are applied. This method is the best one to use on curved or irregular surfaces.

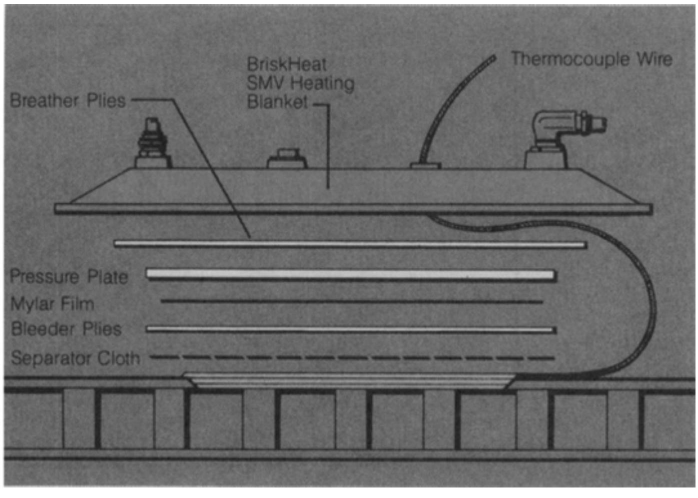


Notes: (1) Precured 3 or 6 ply patch layer
(2) Titanium foil — 0.008 inch or 0.016 inch thick

Fig. 10.3.3 Typical Metal Or Precured Patch Configuration



(a) First generation pad and controller circa 1986-87



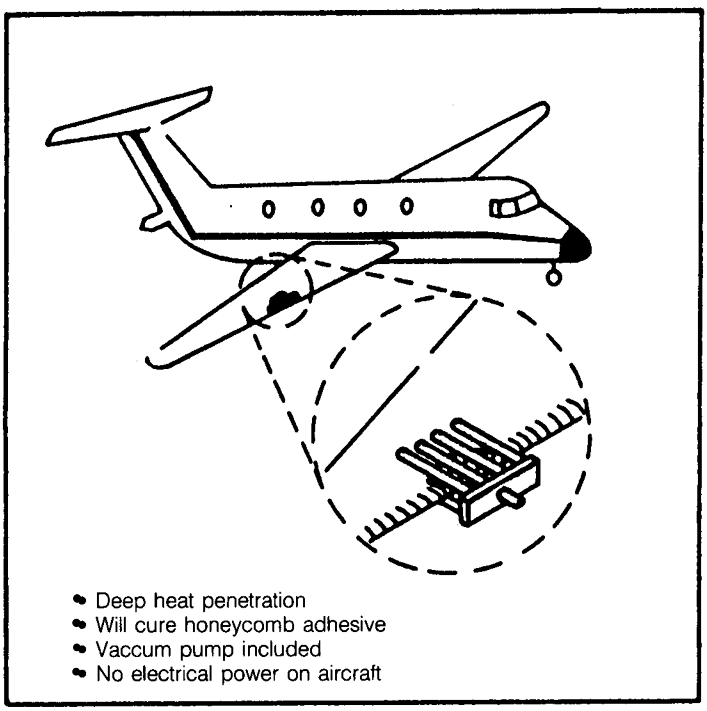
(b) Typical cure layup

By courtesy of BriskHeat Corp.

Fig. 10.3.4 Flexible Silicone Heating Pad

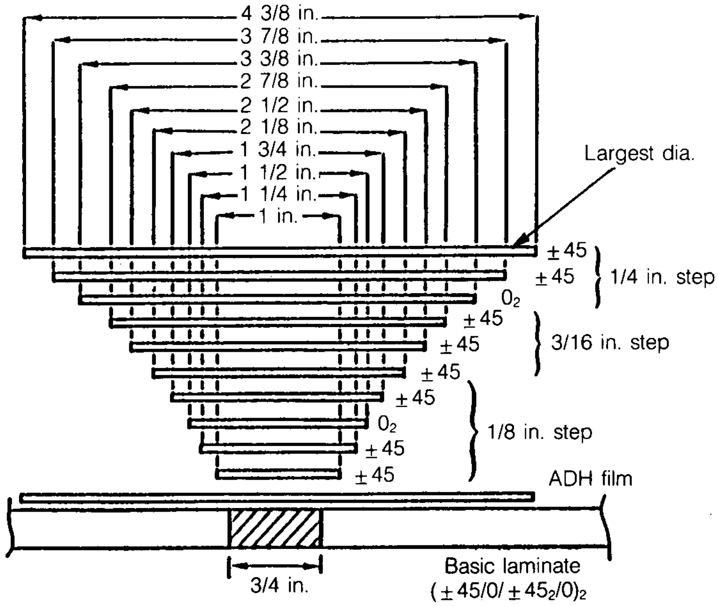
By courtesy of BriskHeat Corp.

Fig. 10.3.5 Second Generation BriskHeat ACR9000 Ramp Programmable Controller, Circa 1990

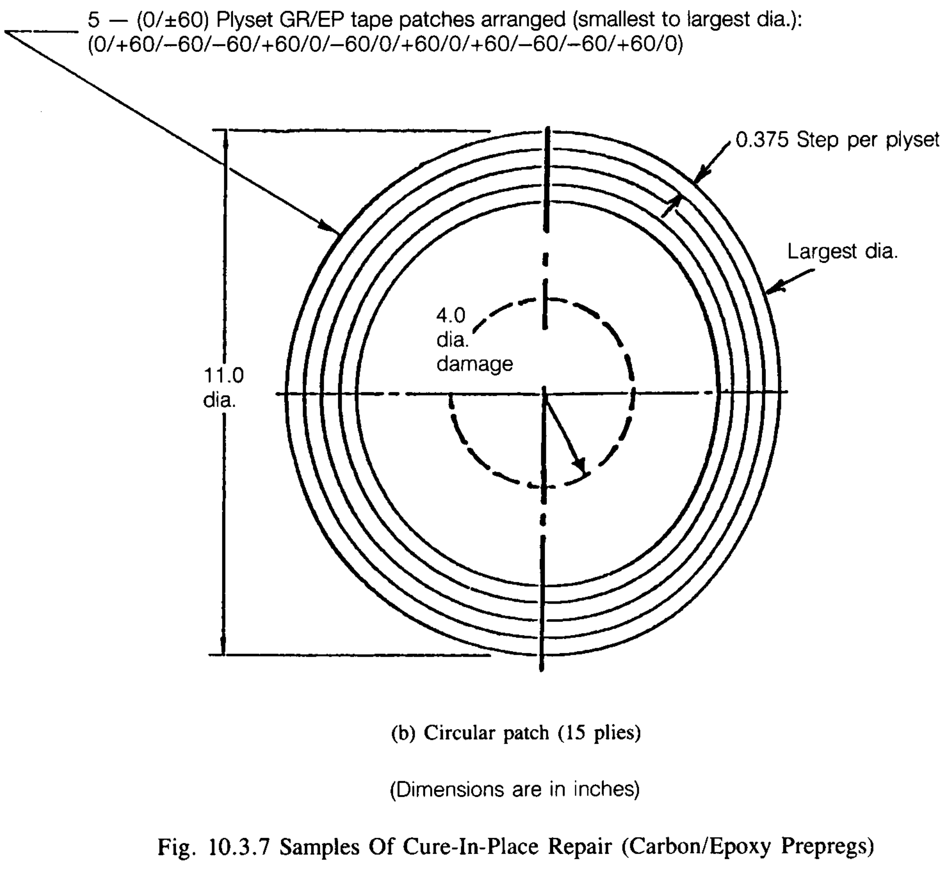


By courtesy of Heat Transfer Technologies

Fig. 10.3.6 Moen Systems (Up To 1500°F)



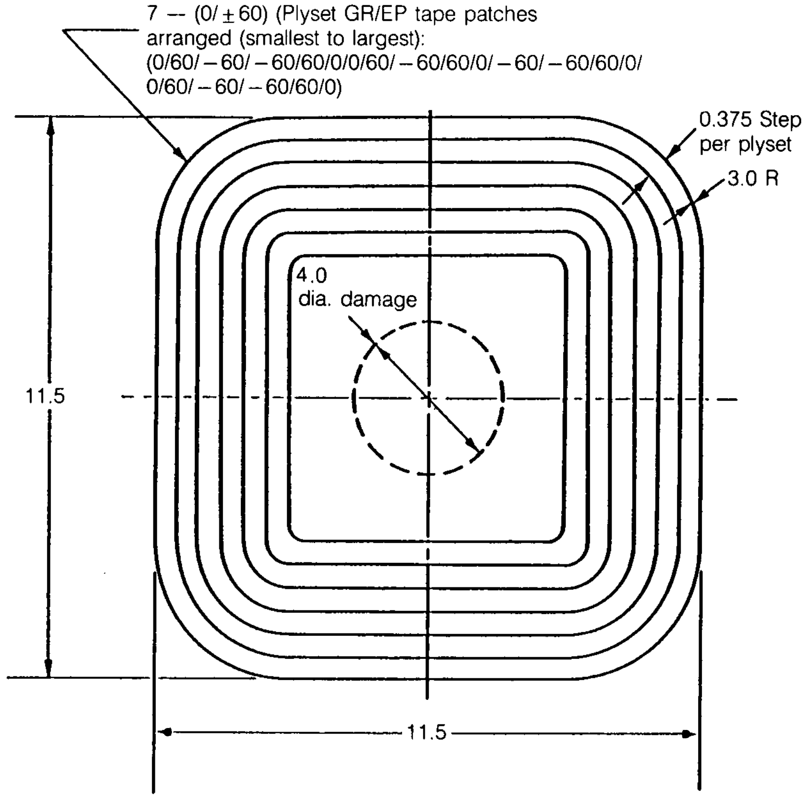
(a) Cut out damaged area (0.75 inch diameter)



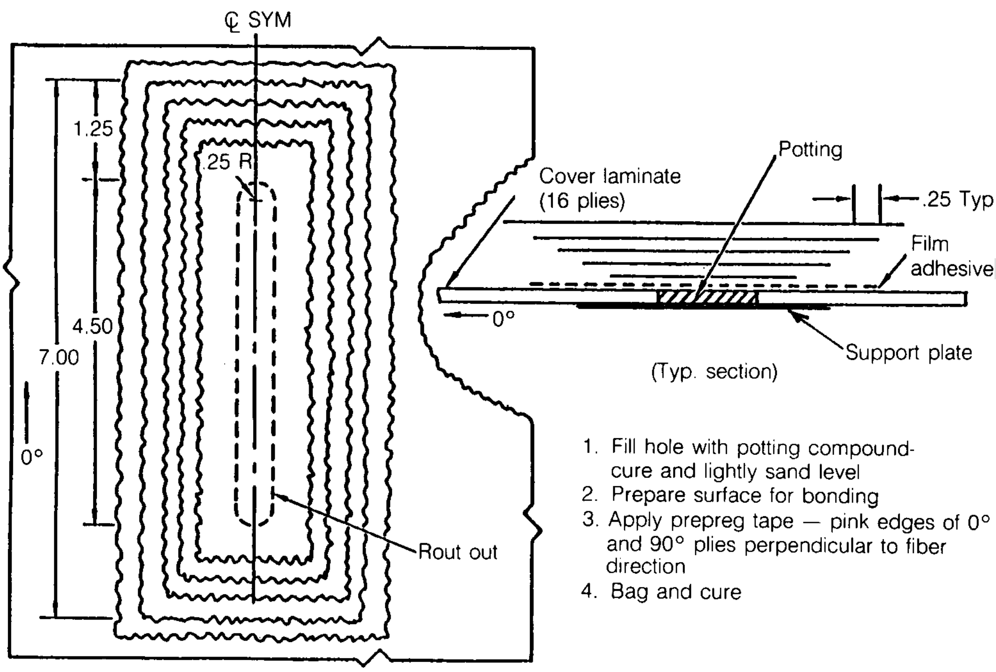
(b) Circular patch (15 plies)

(Dimensions are in inches)

Fig. 10.3.7 Samples Of Cure-In-Place Repair (Carbon/Epoxy Prepregs)



(a) Square patch (21 plies)



(d) Example of repair for a long crack (carbon/epoxy prepreg tapes)

(All dimensions are in inches)

Fig. 10.3.7 Samples of Cure-In-Place Repair (Carbon/Epoxy Prepregs) (cont'd)

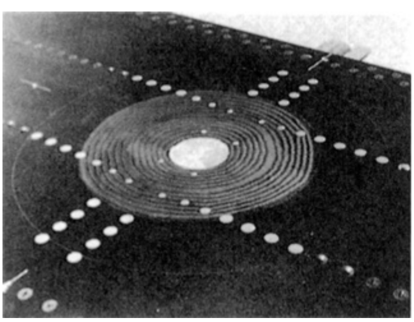
- Fabric prepregs can be readily formed to curved repair surfaces
- Fabric prepreg plies can be readily stepped to form tapers
- Pre-staged prepreg (B-stage prepregs are available and some limited information on C-stage is also available) can be used; pre-staged material has long term storage ability and can still be formed to curved surfaces after storage
- Carefully control to obtain the low level of porosity in a patch when cured under vacuum pressure only

(b) Precuring — Precuring often achieves greater strength than cocuring, but requires more time. A replacement patch can be precured in an autoclave. This method tends to work better using several thin patches, e.g., 3 ply and 6 ply patches on curved surfaces or one thick patch for flat surface
- Patch can be multiple thin layers forming a step-lap or scarf repair (see Fig. 10.1.1) or a single splice plate (for simplicity) with chamfered edges
- This approach can readily utilize pre-kitted patch materials

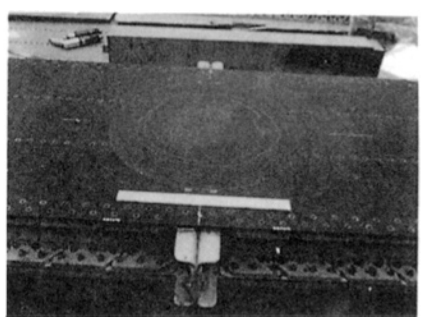
(c) Bonded metal Patch
- One piece aluminum or titanium patch or titanium foil (see Fig. 10.3.3)
- Vacuum bonded or autoclave bonded
- Use of film adhesive or two-part system adhesive

Flush Tapered Bonded Patch

A flush tapered bond patch repair can restore full design strength. This type of repair costs more than other methods and is usually restricted to field level repairs. This method machines the laminate to a scarf surface for greater joining strength, as shown in Fig. 10.3.8; several external plies can be added which overlap onto the repair area. These edges of overlap plies (0° plies) are cut with standard pinking shears to produce serrations 0.125 inch (3.2 mm) deep, as shown in Fig. 10.3.9. The serrations have been shown to prevent peeling of the longer plies, thus resulting in a significant improvement in strength. This method can be used to repair a skin with one or more plies damaged and even through the thickness delamination. This is the most structurally efficient patch approach, but it is expensive and time-consuming.



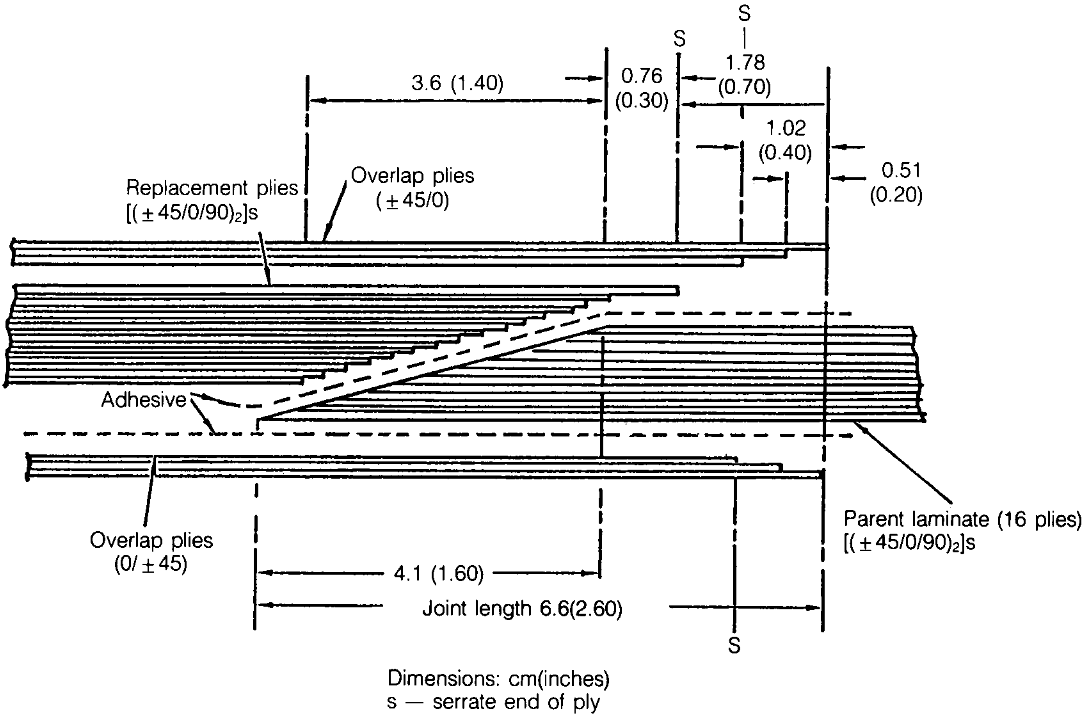
(a) Remove damaged material

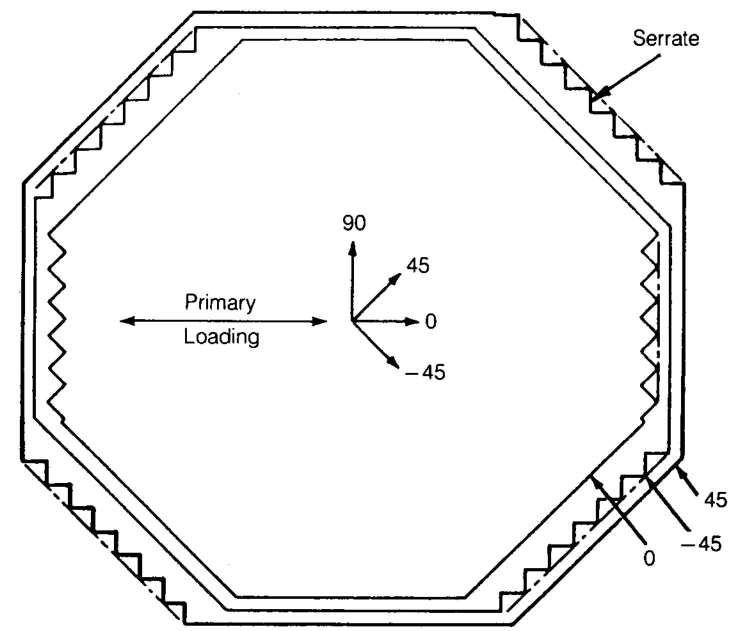


(b) Flush repaired panel

By courtesy of McDonnell-Douglas Corp.

Fig. 10.3.8 Typical Flush Repair (Scarf)





(b) Ovelap serrated plies

Fig. 10.3.9 Examples Of Flash Repair Configuration

Flush repair basically consists of removing the damaged area and replacing the plies:

- Inspect the part to determine the location and extent of the damage
- Remove the damaged area by cutting or sanding to form a scarf surface (sloping 1:30)
- Dry the surface with hot air, a heat gun, or in a oven, etc.
- Orient and stack the plies in the same sequence as the original laminate
- Place a vacuum bag over repair area
- Cure by heating with a heat blanket or similar heating system
- Inspect the repaired area, using ultrasonic inspection methods (see Chapter 9)
- Patch material is usually cure-in-place material with patch orientation matching the part orientation ply-for-ply

When the criteria cited for bolted repair for an external repair patch cannot be met, an alternative scarf or step-lap patch repair is considered. Testing has shown that a scarf repair [see Fig. 10.1.1(d)] is the most efficient joint that can be achieved, and that step-lap repair [see Fig. 10.1.1(c)] in the next most efficient method. However, a scarf joint is relatively easier to machine as opposed to accurately machining/cutting out a step-lap repair. Scarf or step-lap patches are generally considered to be flush patches and normally will have only a few doubler plies external to the moldline. These additional external doubler plies are designed to reduce the peel stresses that will build up at the end of the joint. This type of repair is expensive, requiring extra time and skill. The patches are uniquely designed, and fabricated from prepregs, then cocured and adhesively-bonded on in an autoclave or by the heating blanket curing process. Scarf and step-lapped patch repairs are applied only at the depot or maintenance base.

The most difficult part of the scarf operation is cutting the circular patches to the right size:

- Patch is too small leave gaps which are filled by resin-a weak area
- Patch is too big fold up at the edges

Wet or Prepreg Layup Repairs

Wet layup repairs usually are suitable for lightly loaded structuraes or crack damage and areas not subjected to high temperatures since cure occurs at room temperature. Wet layup patches are uniquely designed from dry woven cloth which has be wetted with resin by hand-impregnation.

These patches have the following characteristics:

- Any number of plies
- Any ply stacking sequence
- Shape specified by the repair engineer
- Surface may be contoured
- Lower strain level (e.g., 2000 μ in/in)

If damage is located in a region containing lower applied strain, multiple surface ply drop-offs, sharp contour breaks, or complex contours, the best repair is a precured "wet layup" patch. If high strain is required, a prepreg patch can be fabricated on a moldform tool or spare part. A vacuum bag is generally required because it allows vacuum and heat to be applied at the repair area.

10.4 HONEYCOMB REPAIRS

Honeycomb panel damage is usually found visually or during NDI. However, with honeycomb core damage it is often difficult to ascertain the full extent of the damage until the skin over the damaged area has been removed.

Repairing honeycomb panels frequently entails cutting out a new piece of core to replace the old and splicing it firmly in place with foaming adhesives. Exterior skin plies are then built up in configurations which match the original skin. For rapid repair with minimal equipment, simply filling in a small damaged area with a body filler or syntactic foam can return a part to its original aerodynamic surface profile frequently with little or no performance penalty, and only a small weight penalty.

The usual course of action for a very badly damaged honeycomb panel is to carefully remove the damaged portions of skin and core and sand away any paint and/or primer which may have been exposed. Replacement plies are cut out to exactly match the damaged area, and are laid up in the same orientation as the original prepregs. The repaired area is then covered with a vacuum-bag to apply pressure to the area during cure for maximum bond integrity and minimal voids.

A common repair problem with honeycomb panel structures is ingression of moisture into the honeycomb, usually as a result of microcracking in the composite skins. Skins must be peeled back, the honeycomb core dried out or replaced, and a repair patch applied. If moisture can get into the honeycomb core, it will become steam during the repair heating procedure, and can blow the part apart.

The type of repair used depends upon the type and extent of the damage, as well as on the loads in the area. The amount of repair weight which can be added to the area may also be a factor. Honeycomb panel can be repaired by following methods:

(a) Method A — For minimal surface damage:
- Clean up the damage surface
- Fill in the damaged area with filler or syntactic foam

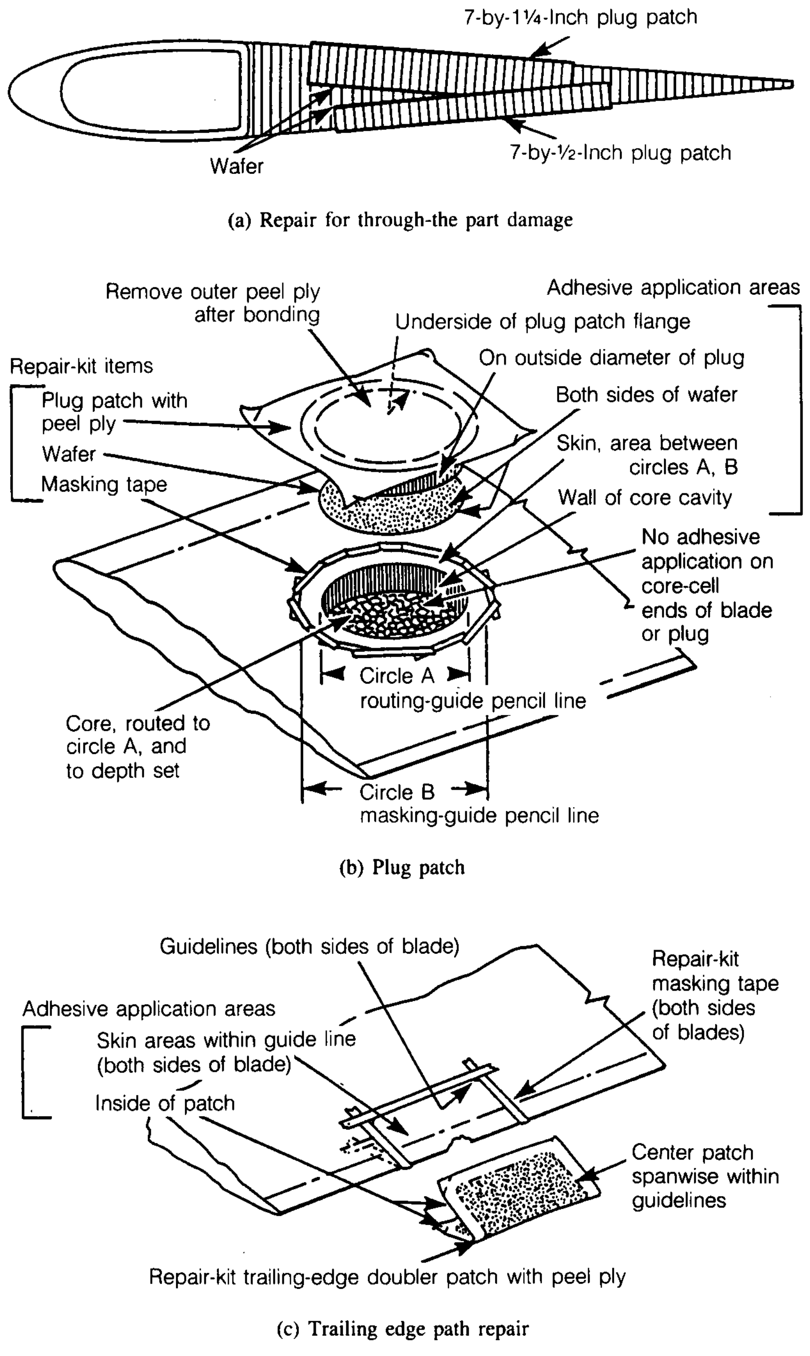
(b) Method B — For intermediate damage:
- Use potting compound to stabilize the parent core, if:
  - Damage area is small, and/or the loads are not high
  - The repair weight does not exceed the allowable weight limitation
- If the limitations cited above are exceed then the following Methods C and D should be considered

(c) Method C — Use syntactic foam filler:
- Remove damaged area
- Fill in with syntactic foam or equivalent
- Filler(s) are sanded flush with the surrounding skin
- Bond or cocure the external skin over it

(d) Method D — Use honeycomb filler:
- Remove damaged area
- Fill in with new piece of honeycomb core
- Use adhesive or syntactic foam to hold the new core in place
- Bond or cocure the external skin over it

Fig. 10.4.2 shows how composite helicopter blades can be repaired in the field:

- A plug patch corrects skin and core damage [see Fig. 10.4.2(a)]
- A double-plug patch repairs holes [see Fig. 10.4.2(b)]
- A V-shaped patch is applied to the trailing edge [see Fig. 10.4.2 (c)]

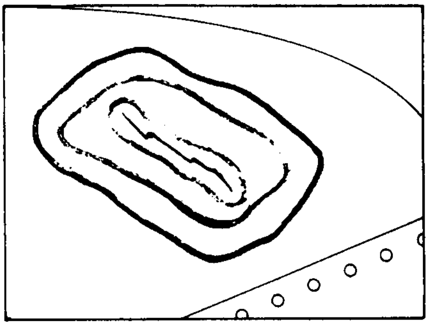


By courtesy of Kaman Aerospace Corp.

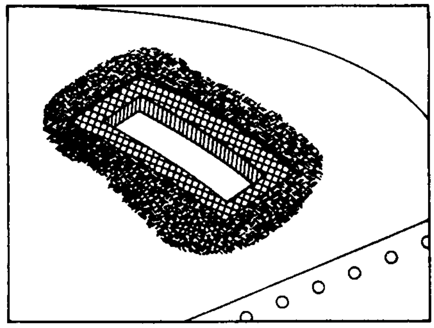
Fig. 10.4.2 Honeycomb Blade Repair

Example of Field Fixes for Beech Starship honeycomb panel

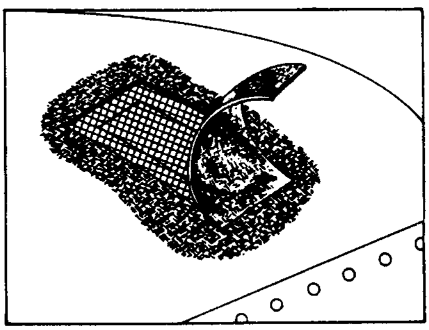
For the all-composite Beech Starship, a special repair method which utilize pre-cut prepreg patches and other special materials was formulated as the airframe was designed. It specifies all the necessary laminates, the ply sequences, and orientation of fibers.



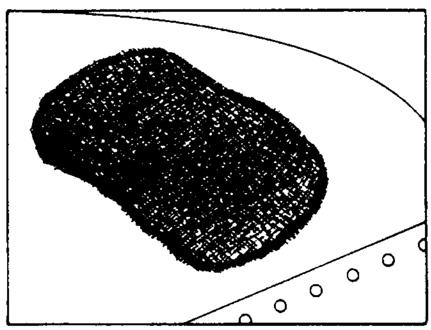
(a) Areas are marked to define actual damage and to locate area to be cleaned and prepared for patching process.



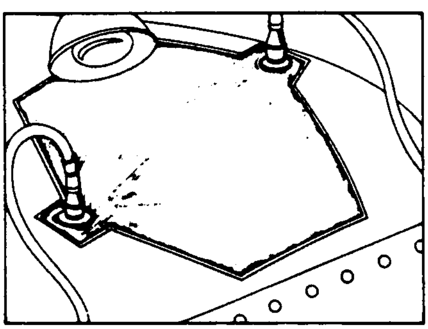
(b) Damagaed area is removed to evaluate inside damage and to make further preparations for patching.



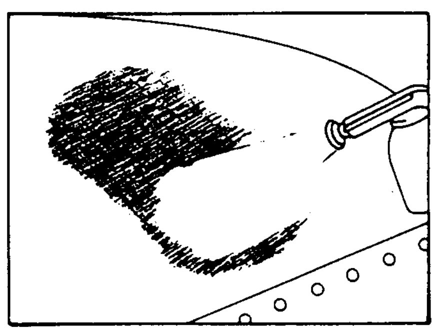
(c) The damaged core is replaced and epoxy painted graphite is applied in the same orientation and number of plies as the damaged skin which has been removed.



(d) One or two plies of graphite/epoxy are added on top of patch to make a "bridge" to undamaged area in order to accommodate designed load.



(e) A vacuum bag is applied to repaired area to assure good contact during 24 hours of curing. Then the repair is subjected to one hour of heat at 180 degrees F.



(f) Finally, the repaired area is sanded, primed and given a final coat of paint. It's as good as new!

By courtesy of Beech Aircraft Corp.

Fig. 10.4.3 Example of A Field Fix for A Beach Starship Skin

The repair of a Starship skin panel follows a controlled order which is specified in the detailed procedure. In the example shown in Fig. 10.4.3, the repair process involves a load bearing area of a wing skin punctured by a large tool which was dropped from a considerable distance. The drop splinters the fibers and penetrates the skin, crushing part of the Nomex core.

The major steps in the repair procedure are as follows:

(1) The first step is to inspect the extent of the damage. This is typically done by tapping the laminate and listening for resonant changes in the sound. A circle is then drawn around the area to define the actual damage.

A second circle is drawn beyond that one to indicate the area which will help support the patch. And then a third circle on the outside of the second will define the total area to be cleaned and prepared for the patching process.

(2) The second step is to sand the surface and wipe it down with a special solvent to assure that it is bondable.

(3) The third step is to remove the face sheet over the damaged area and replace any damaged core.

(4) The fourth step is to lay on as many plies of graphite, painted with epoxy, as are necessary to fill the gap left from the removal of the damaged face sheet. This must be the exact number of plies as contained in the original skin and the plies must have the same orientations as the original skin.

(5) The fifth step is to build a bridge over the damaged spot to undamaged areas using woven graphite fiber soaked in epoxy. Enough layers are used to create a patch that restores the ability of the part to carry its designed load.

(6) The plies laid down are carefully, feathered on their outer edges to make a smoother patch.

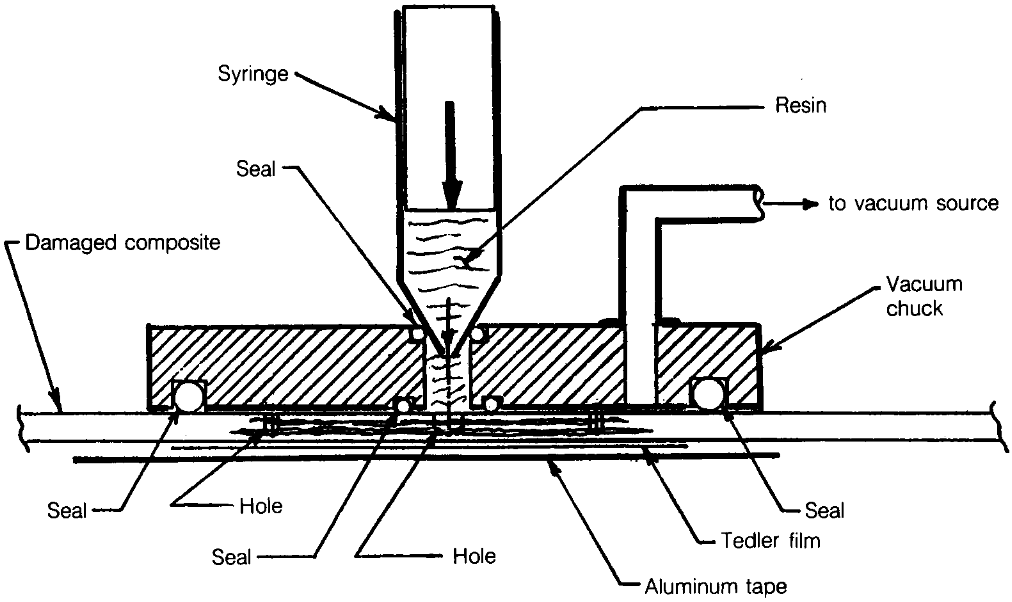
(7) The final stage of the repair involves covering of area with a vacuum bag (a depressurized plastic sack) to assure good contact during curing.

(8) When the full epoxy reaction has taken place, the repaired area is ready for sanding, priming and painting.

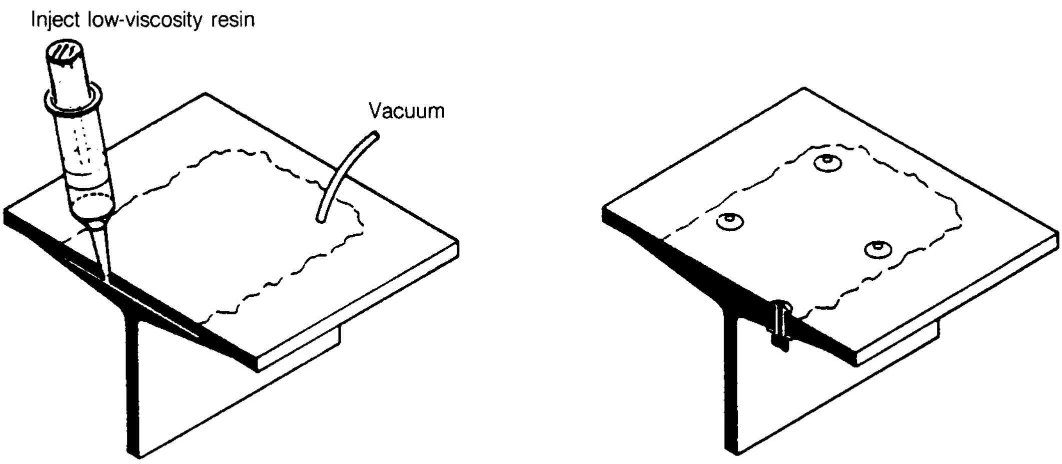
(9) The whole process is simple and can usually be done overnight.

10.5 INJECTION REPAIRS

This method, as shown in Fig. 10.5.1, injects resin directly into the delaminated area, without removing the damaged materials. Resin injection can be performed quickly and right in the field. The resin injection method of repair is the primary approach for repairing delaminations and skin-to-core disbonds. Resin is injected into the damaged area at an edge, or through drilled holes/fastener holes, and re-glues the damaged area together. Some delaminations can be repaired by drilling two holes through the skin to the depth of the delamination, and then injecting resin into one hole until it flows freely from the other hole.



(a) Setup for resin injection



(b) Bonding pressure by using fastener's clamped up force

Fig. 10.5.1 Resin Injection Method

Injection repair for filling of internal delaminations identified by NDI is a uniquely composite repair, and is not analogous to any type of metal repair. This repair can be readily accomplished at depot level or even a field level facility, using a two-part low viscosity resin.

Generally, injection repair is not effective for filling internal delamination in GR/EP structure, because it will not flow enough to create a bond.

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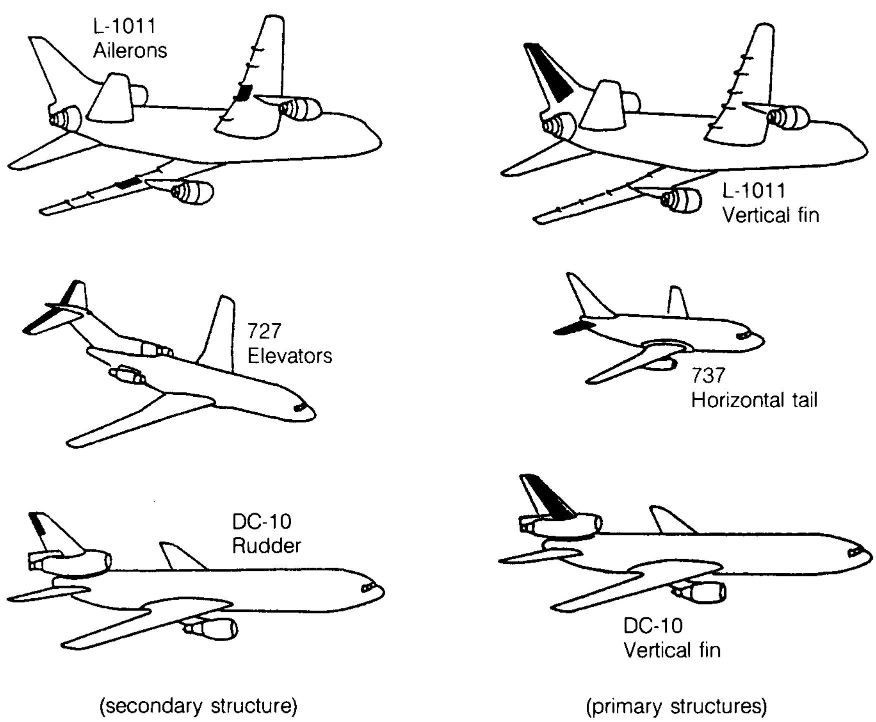
Chapter 11.0

COMPOSITE APPLICATIONS

11.1 INTRODUCTION

Composites are now routinely used on aircraft and not only on control-surfaces and fairings but also on primary structures on both military and commercial aircraft. It can only be a matter of time before aluminum alloys are ousted from their position as the primary airframe material. Composites such as carbon fiber-, Kevlar-, and fiberglass-reinforced laminates have made progressive inroads into airframe designs and some manufacturers of business aircraft have embraced composites wholeheartedly. Much of the credit for persuading manufacturers to obtain composite experience goes to NASA (see Fig. 11.1.1), which in the 1970s spent more than $60 million on the Aircraft Energy Efficiency (ACEE) Program for the design, manufacturing, and testing of composites.



(a) Aircraft components of ACEE program

Fig. 11.1.1 NASA Aircraft Energy Efficiency (ACEE) Programm

Aircraft component	Planform area, m^2	Chord		Span, m	Metal Design weight, kg	Composite Design weight, kg	Weight savings %
		root, m	tip, m				
Secondary Structures:							
727 Elevator	4.1	1.24	0.53	5.26	128.7	95.7	25.6
DC-10 Rudder	3.0	.97	.60	4.00	41.4	30.3	26.8
L-1011 Aileron	3.2	1.34	1.39	2.49	64.1	47.3	26.2
Primary Structures:							
737 Horizontal stabilizer	4.8	1.31	0.61	5.09	118.2	86.2	27.1
DC-10 Vertical fin	9.36	2.07	1.10	6.95	456.0	363.9	20.2
L-1011 Vertical fin	13.9	2.73	1.31	7.62	390	283.2	27.4

(b) Structural data and weight savings

Fig. 11.1.1 NASA Aircraft Energy Efficiency (ACEE) Program (cont'd)

Product	Component and/or producer	Composite
	PRODUCTION	
F-14	Horizontal stabilizer	Boron-epoxy
F-15	Tail section, cabin floor, and stabilator	Boron-epoxy
UTTAS Helicopter	Structural beam reinforcement	Boron-epoxy
F-111	Wing pivot doubler	Boron-epoxy
Mirage 2000	Rudder	Boron/graphite-epoxy
Space shuttle	Fuselage	Boron-aluminum tubes
	RESEARCH AND DEVELOPMENT	
F-14	Overwing and fairing	Boron + other filaments/fibers-epoxy
A-7	Outer wing	Boron/graphite-epoxy
C-130	Wing box	Boron-epoxy-reinforced aluminum
F-4	Rubber	Boron-epoxy
B707	Foreflap	Boron-epoxy
F-100	Engine fan-blades	Boron-aluminum
C-5A	Wing slat	Boron-epoxy
CH-54	Fuselage stringer and tail skid	Boron-epoxy
B-1	Horizontal and vertical stabilizers and wing slat	Boron/graphite-epoxy
F-111	Horizontal stabilizer	Boron-epoxy
CH-47	Rotor blade	Boron-epoxy

Fig. 11.1.2 Early Military Boron Composite Programs

With each new generation of military aircraft, the use of composite materials has increased. Combat aircraft entering service in the 1990s will be the first to be designed from the ground up to take full advantage of composites (see Fig. 11.1.2).

Currently, 10 percent of the structural weight of existing modern combat aircraft, such as the F-16 or F-18, is accounted for by composites. This proportion is only a few percent for the latest commercial transport, but the tendency is towards more widespread use of composites. The aim of second-generation composite development is to lower manufacturing cost while improving the acceptable stress limits, particularly in compression. Good damage tolerance properties and acceptable performance of the matrix-fiber combination in environmental conditions, such as those encountered in operation must be maintained.

The following composite applications are for reference only; some are in production and others are only in development. This chapter does not cover all composite applications but a few select components.

11.2 COMMERCIAL TRANSPORT AIRCRAFT

Application of composites in commercial transports has generally lagged behind military usage because:

- Cost is a more important consideration
- Safety is a more critical concern, both to the manufacturer and government certifying agencies
- A general conservatism because of past experiences with financial penalties from equipment down time

Composite materials are widely used, not only for exterior secondary components such as leading and trailing edges, fairings, radomes, landing gear doors, etc., but also for fuselage interiors on commercial transports. All interior materials must meet both flammability resistance (if applicable) and smoke and toxic-gas emission requirements. In general, the phenolic resin system is used for aircraft internal applications because of its excellent fire-resistant properties. The main considerations for interior panel application are:

- Impact resistance (mainly floor and side-wall panels)
- Stiffness
- Surface smoothness and appearance

NASA Aircraft Energy Efficiency (ACEE) Program

The ACEE program which started in 1975, greatly expanded the scope of commercial transport composites applications; it included three secondary and three primary structures:

(a) Secondary structures
- Lockheed L-1011 inboard aileron
- Boeing 727 elevator
- McDonnell-Douglas DC-10 rudder

(b) Primary structures
- Lockheed L-1011 vertical fin box
- Boeing 737 Horizontal stabilizer box
- McDonnell-Douglas DC-10 vertical fin box

The principal criteria applied to the design of the composite components in the ACEE program were:

- The part must meet the same design load requirement as the present metal part (in some cases, detail parts are being made to match the strength of the metal part even if it was originally over designed)
- The loads induced into adjoining structures must be the same as, or less severe, than the present loads (that is, no modifications of adjoining structure should be required)
- The interface should require little or no change and any change must accommodate a standard metal part as well as the composite part
- Handling characteristics of the aircraft should not be significantly altered (particularly, there should be no change in control response or adverse changes to the flutter envelope of the aircraft)

(1) Lockheed L-1011 Inboard Aileron:

The L-1011 aileron [see Fig. 1.1.3(a)] is a two-spar, five rib-stiffened structure with sandwich skin construction. Because the aileron is located just aft of the wing engines, the aileron must withstand acoustic loadings.

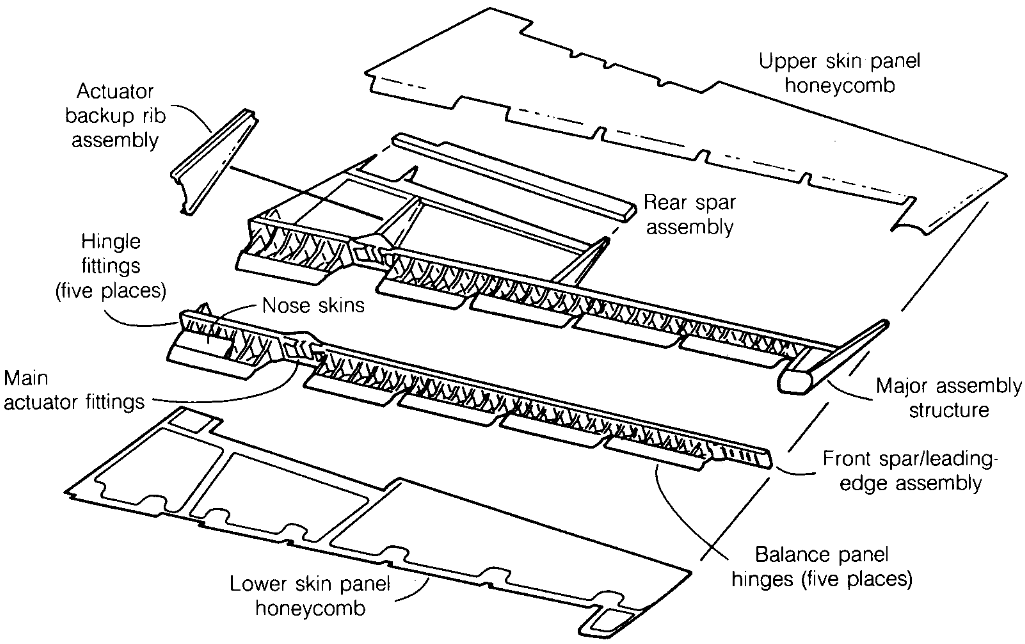
- Skins are made of graphite/epoxy tape facesheets and syntactic foam cores. The syntactic core or Syncore (see Fig. 2.6.14), consists of epoxy resin filled with small glass microballoons
- Graphite/epoxy tape is used for the main ribs; graphite/epoxy cloth for the intermediate ribs
- Graphite/epoxy tape and graphite/epoxy cloth are used for the front spar

(2) Boeing 727 Elevator:

- The Boeing 727 elevators (see Fig. 11.2.1) are constructed using Nomex honeycomb sandwich panels with graphite/epoxy facesheets for surface panels and four ribs.
- Use of sandwich covers allows elimination of most ribs.
- The ribs have honeycomb stabilized webs, and the spars are solid laminates.
- Skin panel facesheets have a layer of graphite fabric oriented at 45° and a single layer of unidirectional tape at 90°. The tape is used as the outer layer of the exterior facesheet to provide a smooth, nonporous surface.
- The outer layer of the inner facesheet is fabric, because it is more resistant to fiber breakout during drilling. Only fabric is used for the ribs
- Due to the weight savings on the elevator, removal of mass from the balance weight contributes to an overall weight savings of 29 percent (25.6% structural savings, see Fig. 11.1.1)

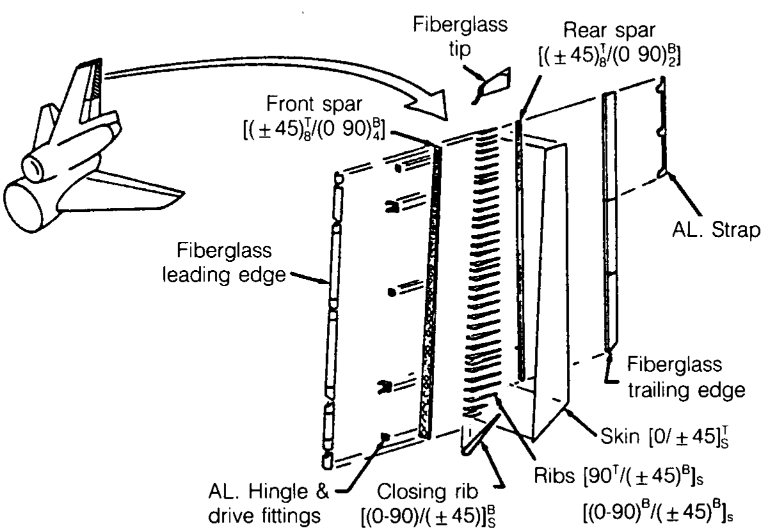
(3) McDonnell-Douglas DC-10 Rudder:

The all-graphite/epoxy structural box (see Fig. 11.2.2), providing a weight savings of 33% (including balance weight), is a two-spar multi-rib construction with solid skins. Skins are made from unidirectional broadgoods and substructures from fabric. Leading and trailing edges are fiberglass. The cost-effectiveness of the design results from following factors:



By courtesy of The Boeing Co.

Fig. 11.2.1 Boeing 727 Composite Elevator-Structural Arrangement



(a) Construction of DC-10 composite rudder. Superscript T denotes uniweave, wide tape and B denotes biwoven fabric. Numerical subscripts denote the number of plies and subscript S denotes symmetry.

Fig. 11.2.2 DC-10 Graphite/Epoxy Composite Rudder

By courtesy of McDonnell-Douglas Corp.

(b) DC-10 Composite rudder

Fig. 11.2.2 DC-10 Graphite/Epoxy Composite Rudder (cont'd)

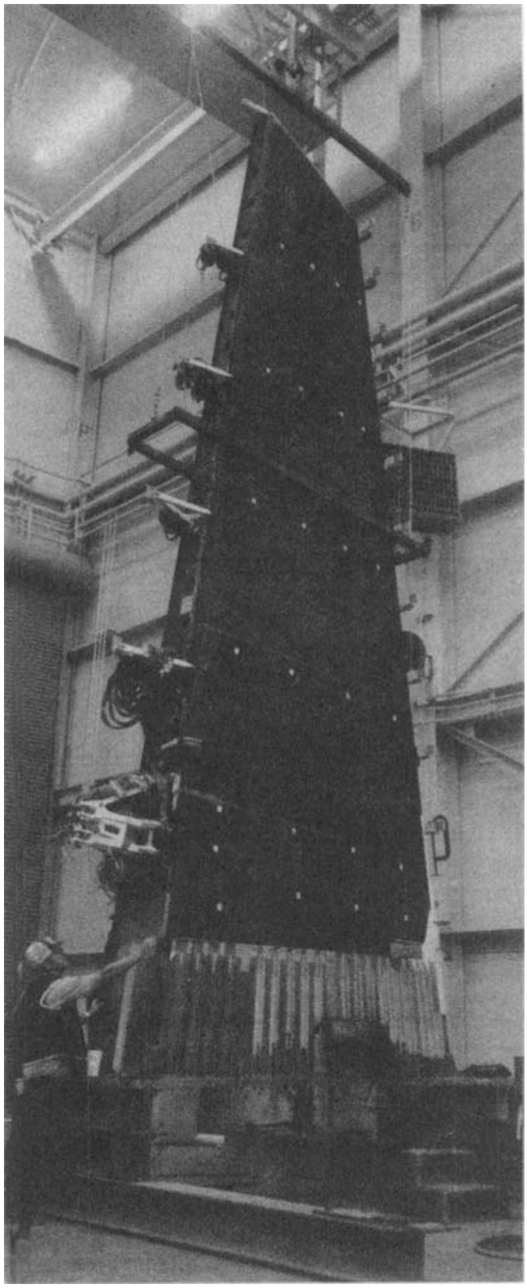
- Use of broadgoods and fabrics, which reduces layup time
- The structural box is made as a single cocured unit using the beanbag approach which uses hollow aluminum beads in an inflatable bag. When a vacuum was pulled on the bag, it forms a solid mandrel which could be used as a form-block for rib channel members
- After assembling the bags within the specified steel curing tool, heat is applied to the bags and the total structure is cured
- Curing is done in an oven which is less expensive to use than an autoclave

(4) Lockheed L-1011 Vertical Fin Box

The composite version of the L-1011 vertical fin box, as shown in Fig. 11.2.3, is composed of solid skins over a substructure consisting of two spars and seventeen ribs. Unidirectional tape was used for the skins on the composite version because of its better mechanical properties and suitability for use with automatic layup processes. The number of ribs was reduced from 17 to 11 when composites were used [see Fig. 1.1.3(b)].

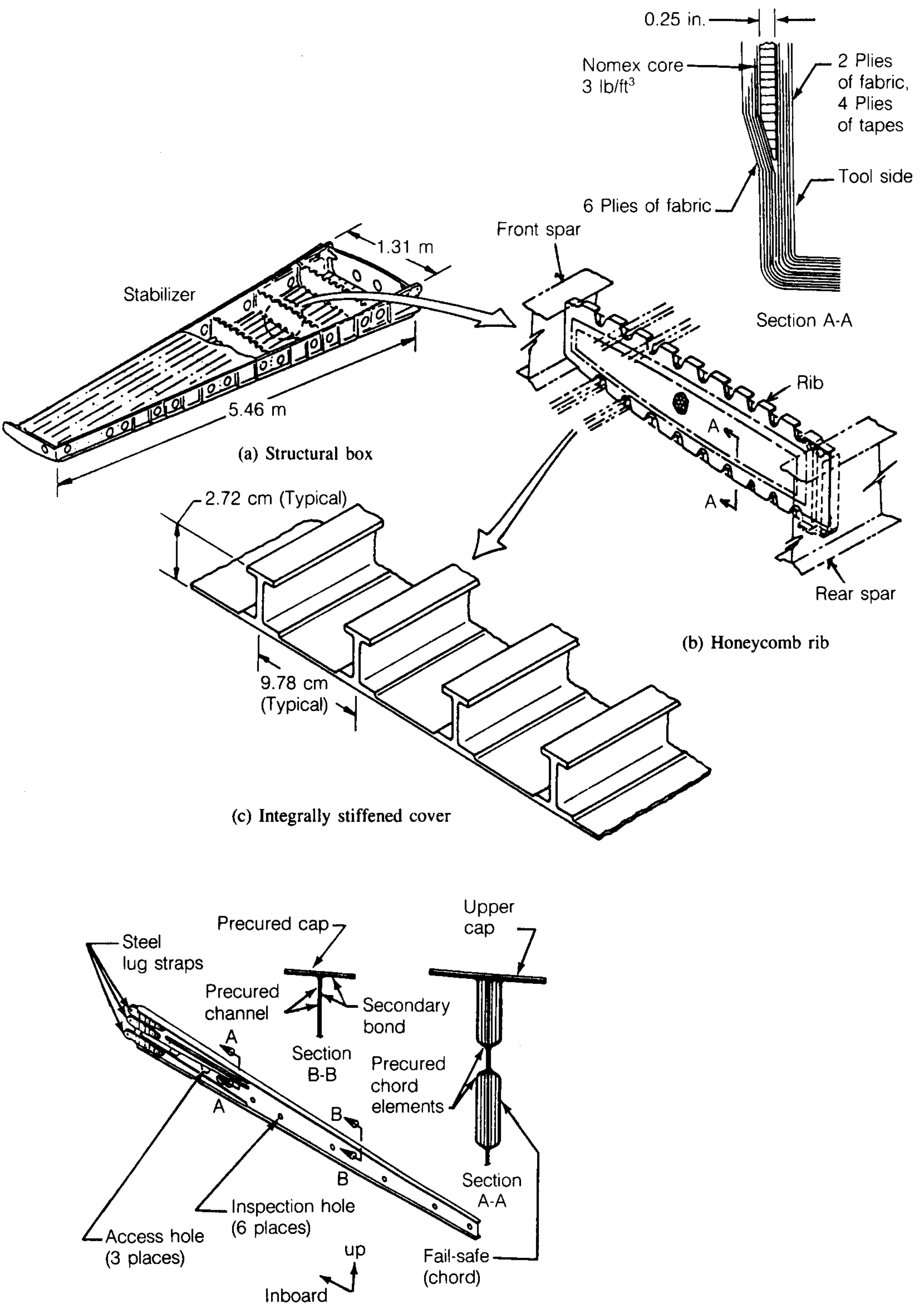
- The three upper ribs were solid graphite/epoxy laminates with integrally molded caps and bead stiffeners
- The eight lower ribs combine graphite/epoxy caps with extruded aluminum truss webs.

- The covers are solid laminates with integral, cocured hat stiffeners which is lighter than a honeycomb design
- The closed hat section was preferred over open channels or I-sections because it does not require tie clips at the ribs to prevent twisting instability



By courtesy of Lockheed Aeronaufical Systems Co.

Fig. 11.2.3 L-1011 Composite Vertical Fin Box



(a) Structural box

(b) Honeycomb rib

(c) Integrally stiffened cover

(d) Spar details

By courtesy of The Boeing Co.

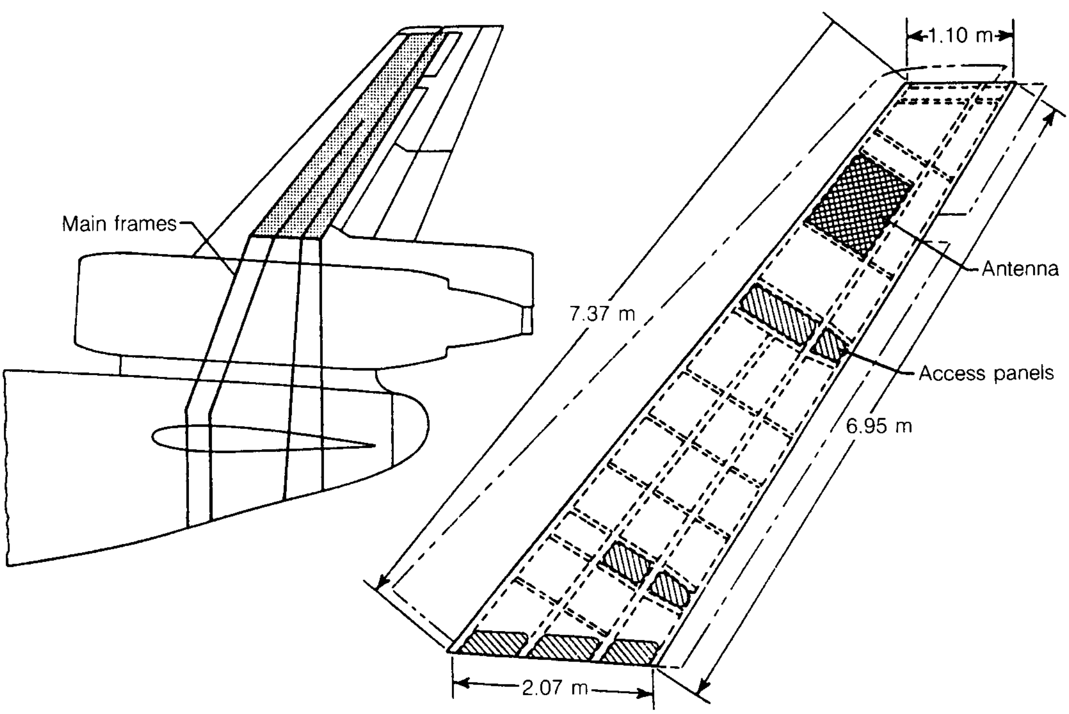
Fig. 11.2.4 Boeing B737 Composite Horizontal Stabilizer

(5) Boeing 737 Horizontal Stabilizer Box:
The Boeing 737 Graphite/epoxy Horizontal Stabilizer (see Fig. 11.2.4) is a two spar, eight rib box construction.

- The cover skin is a solid laminate, I-stiffened skin made from fabric and tape. The cover skin and its integral stiffeners are cocured. The stiffened composite cover skin is only slightly lighter than its unstiffened aluminum counterpart.
- The front and rear spars are solid laminates that are made in two channel sections which are subsequently bonded back-to-back
- The ribs are of a honeycomb sandwich construction in the vertical web portion only; the upper and lower flanges are solid laminates
- The cover panels are mechanically fastened to the substructure
- The major weight reduction comes from the rib and spar, and the reduced numbers of ribs

(6) McDonnell-Douglas DC-10 Vertical Fin Box:
The DC-10 vertical fin box consists of honeycomb sandwich skins with a four spar and thirteen rib substructure (see Fig. 11.2.5).

- The skins consists of fabric graphite/epoxy facesheets oriented at 45° over a Nomex core; two plies on the outer surface and one ply on the inner surface.
- The spars and ribs all use graphite fabric/epoxy sine-wave webs in most areas
- Spar and rib caps are built into the skin sandwich by a gridwork of unidirectional tapes in a quasi-isotropic pattern to locally replace honeycomb.



(a) Structural arrangement

Fig. 11.2.5 DC-10 Composite Vertical Stabilizer

By courtesy of McDonnell-Douglas Corp.

(b) DC-10 Vertical stabilizer

Fig. 11.2.5 DC-10 Composite Vertical Stabilizer (cont'd)

B747,B757, B767 and B777 Transports

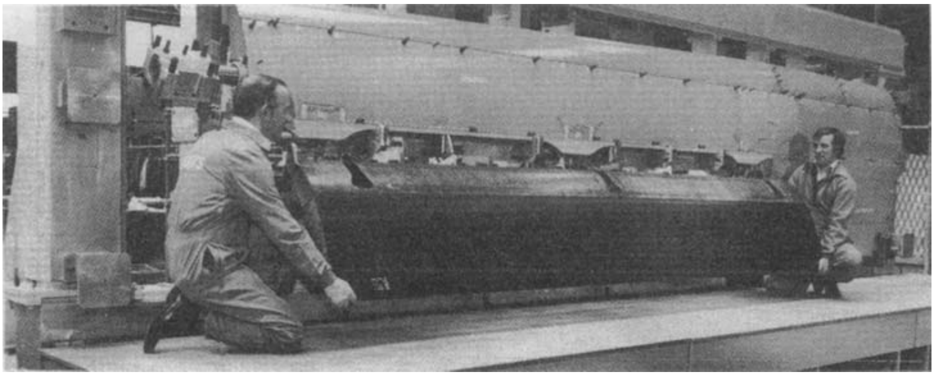
The experience gained from the ACEE programs has resulted in increased composites usage on the next generation of commercial transports, such as the Boeing B757 and B767. In general, with the exception of small, detail parts, most composite components are of honeycomb sandwich construction. Some examples include:

(1) B747 — 6 ft-high winglet, carbon-fiber front and rear spars covered with carbon-fiber epoxy honeycomb sandwich skin panels

(2) B757 — Aileron, inboard spoiler, outboard spoiler, rudder, elevator,and inboard trailing edge flap (see Fig. 11.2.6) are made of carbon/epoxy materials.

(3) B767 — Ailerons, spoilers, rudder, and elevators
- Outboard aileron is a full-depth honeycomb construction, as shown in Fig. 11.2.7
- Rudder cover panels were split for manufacturing reasons. Basic design is a two spar multi-rib box using honeycomb sandwich ribs, spars and cover panels (see Fig. 11.2.8)

(4) B777 — Besides incorporation of composite components used on previous Boeing aircraft [see Fig. 1.1.9(a)], this aircraft has a composite empennage.



By courtesy of The Boeing Co.

Fig. 11.2.6 Boeing 757 Composite Inboard Trailing-edge Flap

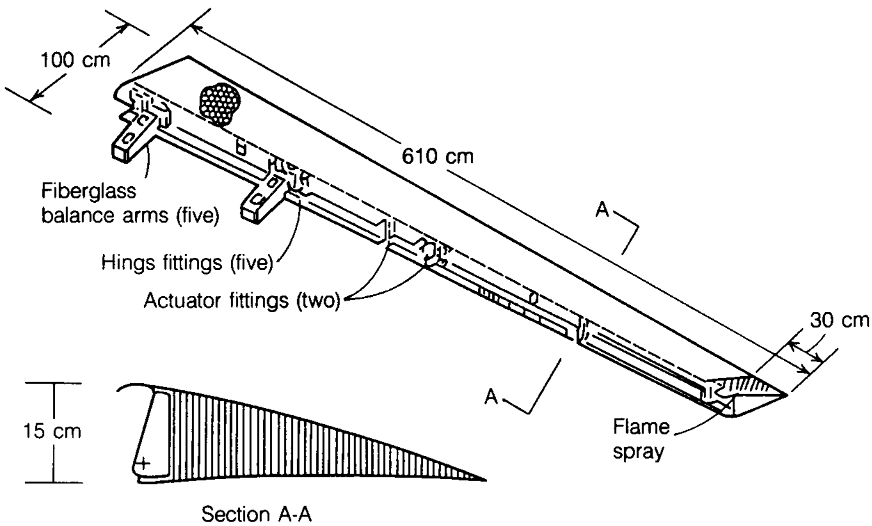
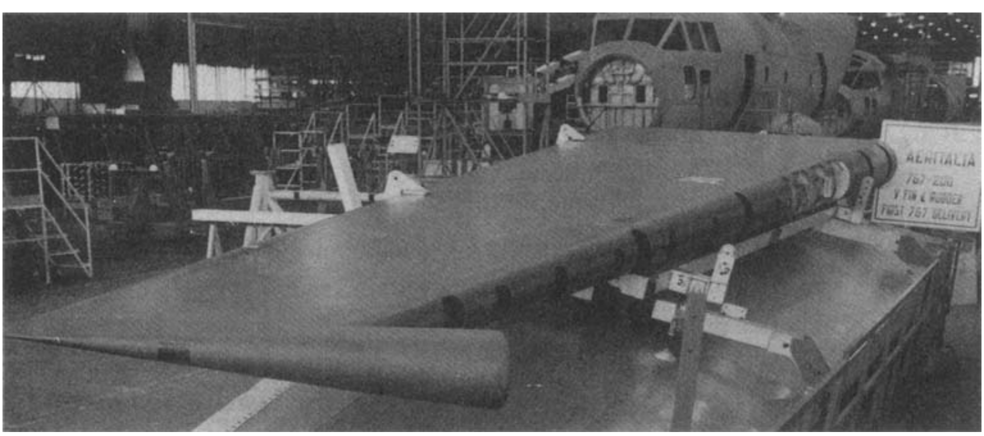


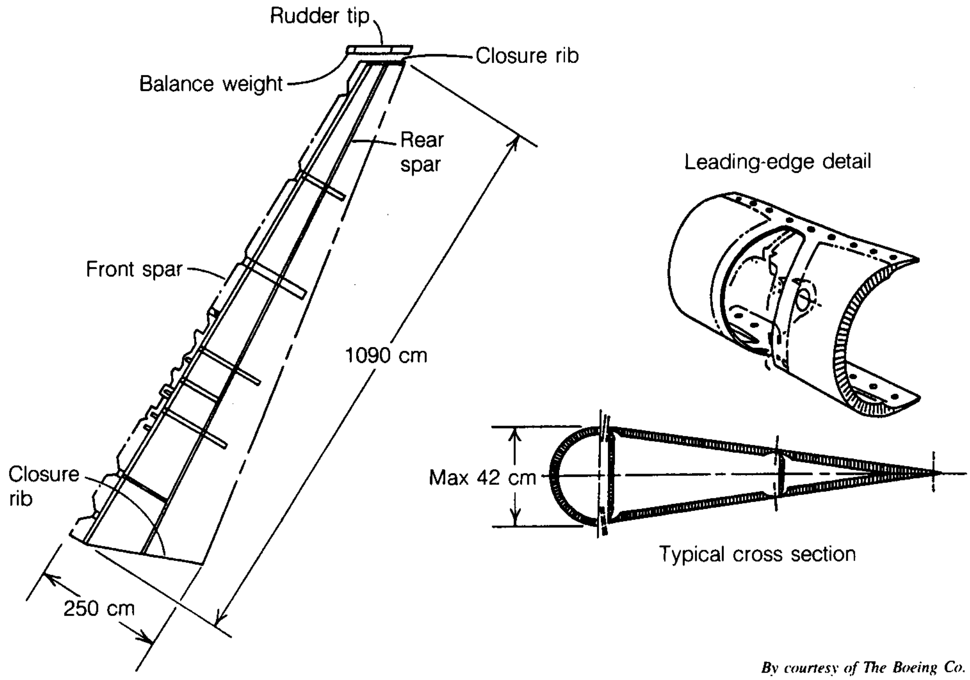
Fig. 11.2.7 Boeing 767 Composite Outboard Aileron



By courtesy of Aeritalia Societa Aerospaziale Italiana S.P.A.

(a) Composite rudder

Fig. 11.2.8 Boeing 767 Composite Rudder



(b) Rudder data and details

Fig. 11.2.8 Boeing 767 Composite Rudder (cont'd)

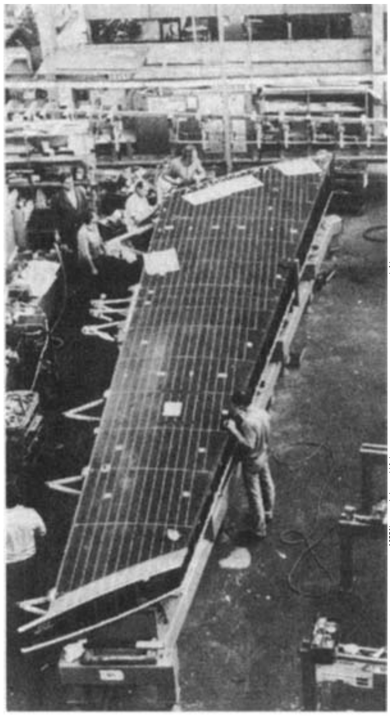
Airbus Transports

In the early design stages of the original A300 transport, the increasing cost of fuel fixed economy of operation by weight reduction as a vital parameter. To further reduce cost of ownership, production costs and maintenance expenditure had to be reduced to the minimum. In 1985, Airbus became the first airframe manufacturer to use composite materials for a series production of primary structures when it began to assemble the A310 with fins built of carbon/epoxy. The use of composite fins reportedly has resulted in cost and weight savings and the elimination of corrosion.

The progress of Airbus composite development is as follows:

- 1972: A300 early design of fiberglass fin leading edge, fiberglass fairings
- 1978-1979: CFRP spoilers, air brakes, landing gear doors, and rudder
- 1980-1985: A300/A310 vertical fin (see Fig. 11.2.9 and Fig. 11.2.10)
- 1985-1987: A320 horizontal tail (see Fig. 11.2.11) and vertical fin using same design approach as A310. One piece module fabrication of A320 flap (see Fig. 11.2.12)
- 1987: Development of new tooling and manufacturing methods in the design of the A330 and A340 vertical and horizontal tail

The A300/A310 vertical fin is a most impressive composite structure. It is 27 ft 3 inches (8.3 m) high and 25 ft 7 inches (7.8 m) wide at the base end represents a weight savings of about 22% compared to its aluminum counterpart. In addition, it consists of only 95 parts compared with 2076 parts in the previous aluminum box structure, insuring a reduction of assembly costs.



By courtesy of MBB GMBH

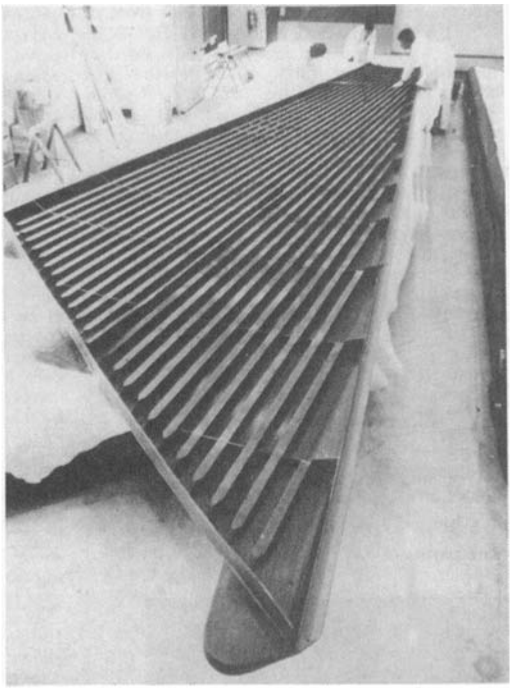
Fig. 11.2.9 Airbus A310-300 Composite Fin In Final Assembly

Production costs are reduced by use of automatic processes for the vertical and horizontal tail boxes. A module concept (Ref. 11.10) evolved and patented by MBB (Germany) is described below:

- Prepreg (usually are ±45°, plies) is wrapped around three-piece- aluminum modules [see Fig. 11.2.10(c)]: this will constitute the web of the stringers and the rib shear tie
- Prepreg is laid into a mold; this will constitute the external skin and stringer bottom flange
- The aluminum modules are assembled into the mold
- Unidirectional tape is laid on the stringer (spanwise) to form the upper flanges
- During the cure pressure is applied to the skin and stringer flanges by the autoclave and to the stringer webs by thermal expansion of the aluminum modules
- A large number of similar modules are set in a fixed grid.
- The complete assembly, under vacuum produced pressure, is cured at 250-350 °F (121-176 °C) and 100 psi (0.69 Mpa) in an autoclave.

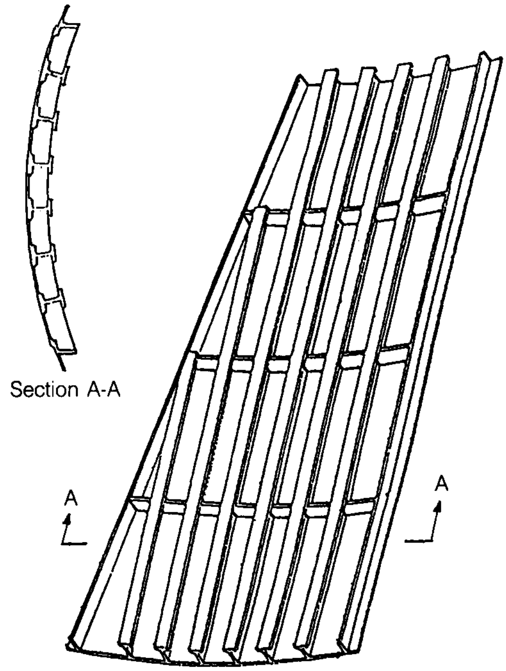
P-180 Avanti

The P-180 designs utilizes composites on the nose cone, forward wing (canard), nacelles, wing trailing edge, empennage (see Fig. 11.2.13), and control surfaces. Composites make up 20% of the total weight.



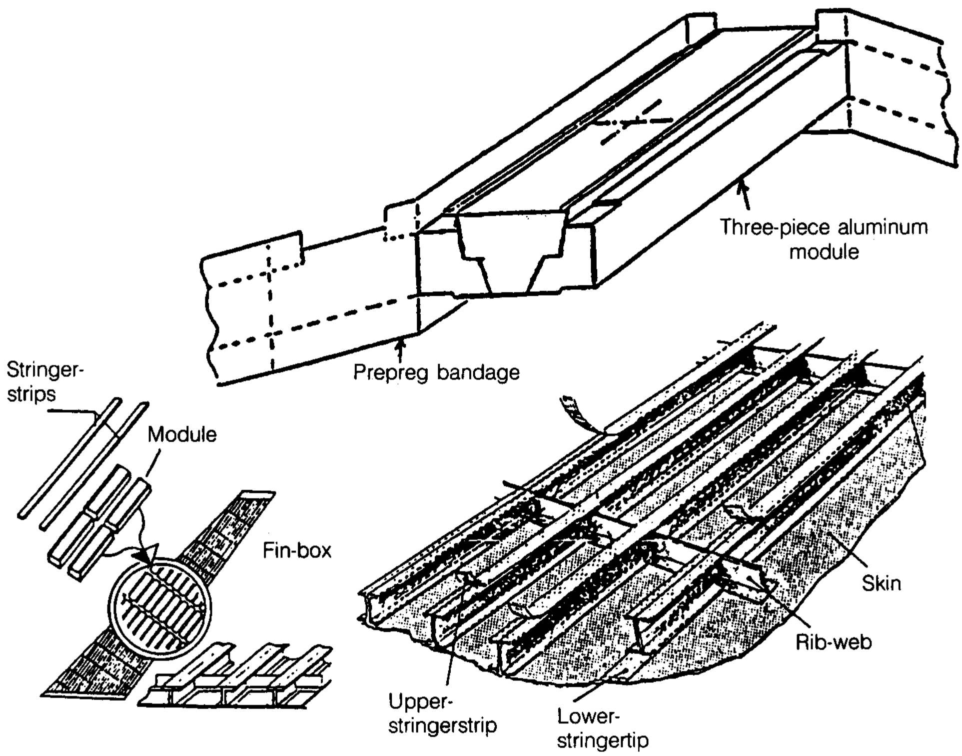
By courtesy of MBB GMBH

(a) Fin skin with integral stiffeners



By courtesy of C.A.S.A.

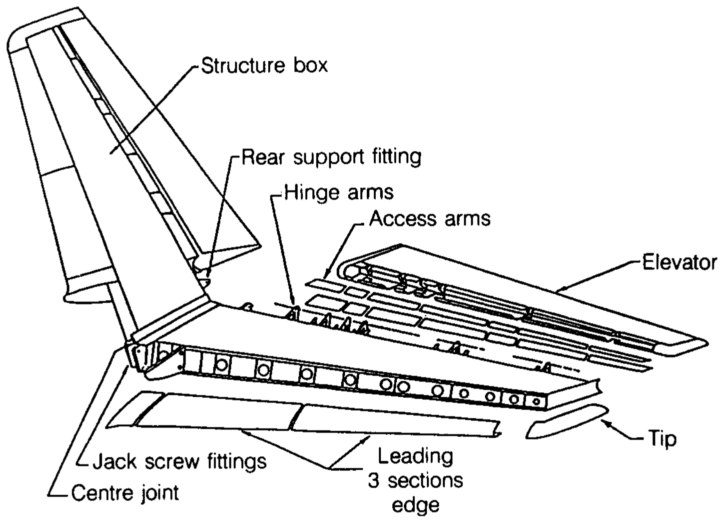
(b) Schematics



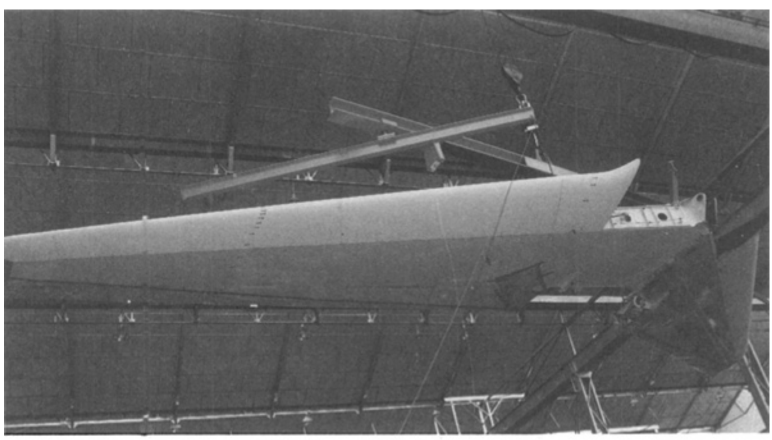
(c) Module usage

By courtesy of C.A.S.A.

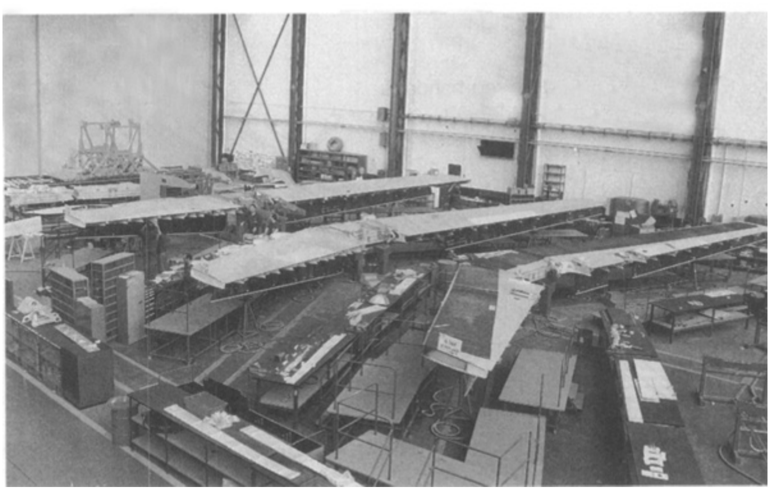
Fig. 11.2.10 Airbus A310-300 and A300-600 Composite Vertical Fin (Skin with Integral Stringers, Spar Caps and Shear Ties)



(a) A320 composite parts



(b) A320 composite horizontal tail



(c) A340 composite horizontal tail

By courtesy of C.A.S.A.

Fig. 11.2.11 Airbus A320 Composite Horizontal Tail

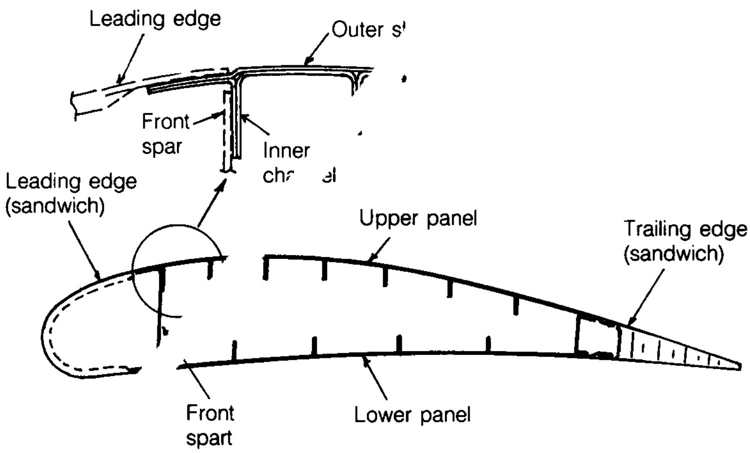
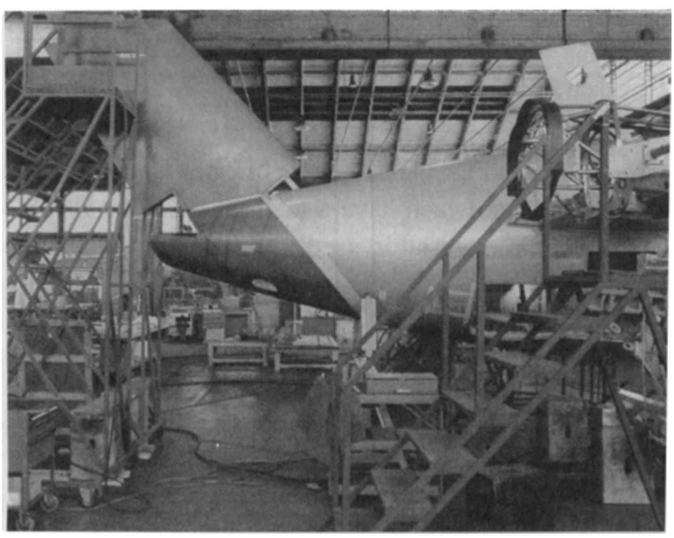


Fig. 11.2.12 Airbus A320 Composite Flap

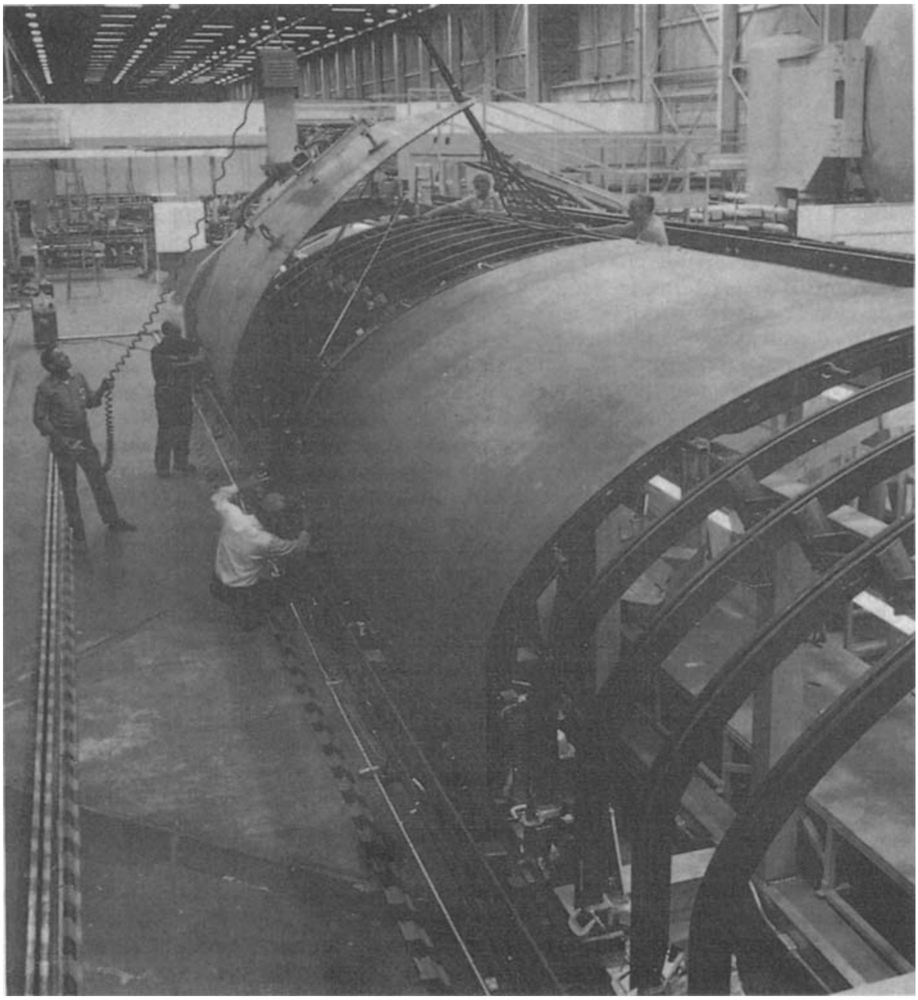
(a) P-180 aircraft



(b) Composite empennge

By courtesy of Rinaldo Piaggio S.P.A.

Fig. 11.2.13 All-Composite Tail of The P-180 Aircraft



By courtesy of Rockwell International

Fig. 11.2.14 Space Shuttle Payload Doors On Assembly Fixture (60ft × 15ft)

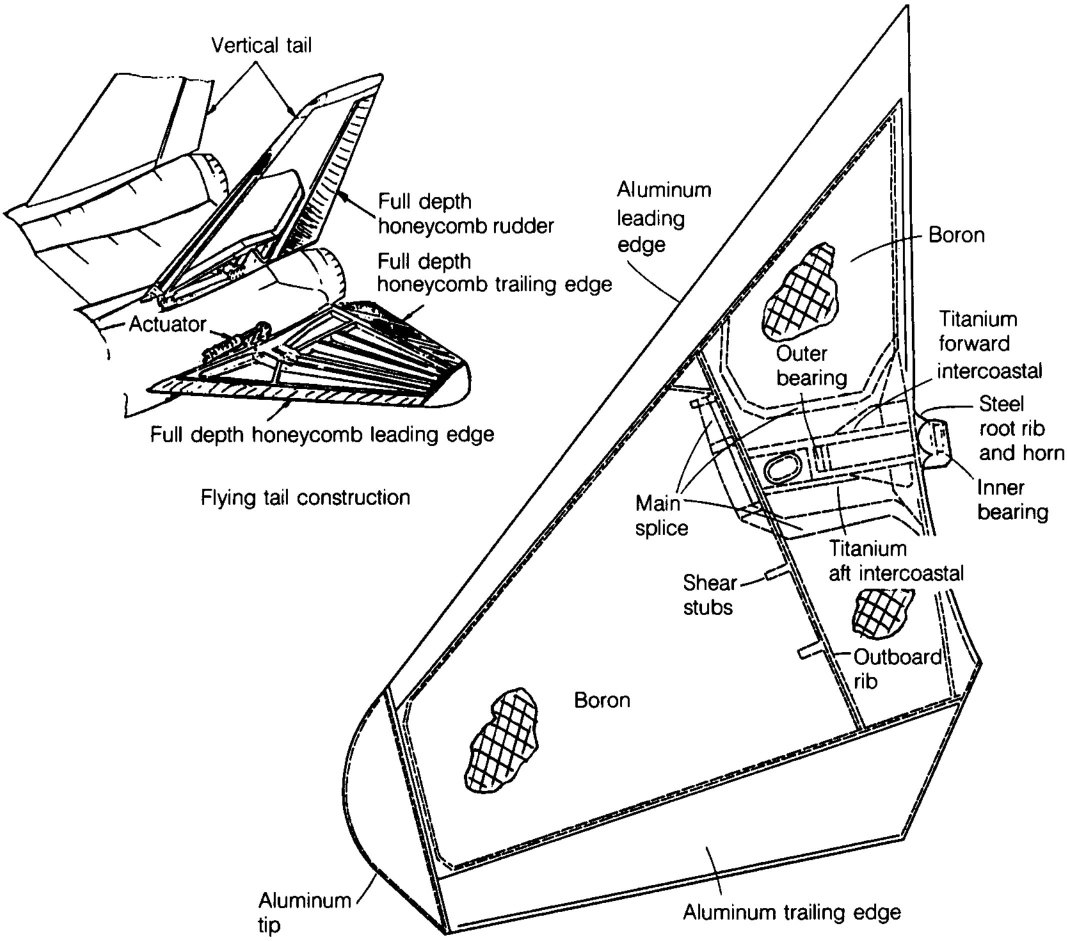
Space Shuttle Orbitor

The upper cargo doors, shown in Fig. 11.2.14, are fabricated from graphite/epoxy materials. The double doors (two each side) each have a total length of 60 ft (18.3 m) and are 15 ft (4.56 m) wide. Honeycomb material is used in the door panels while the frames are made of solid laminate.

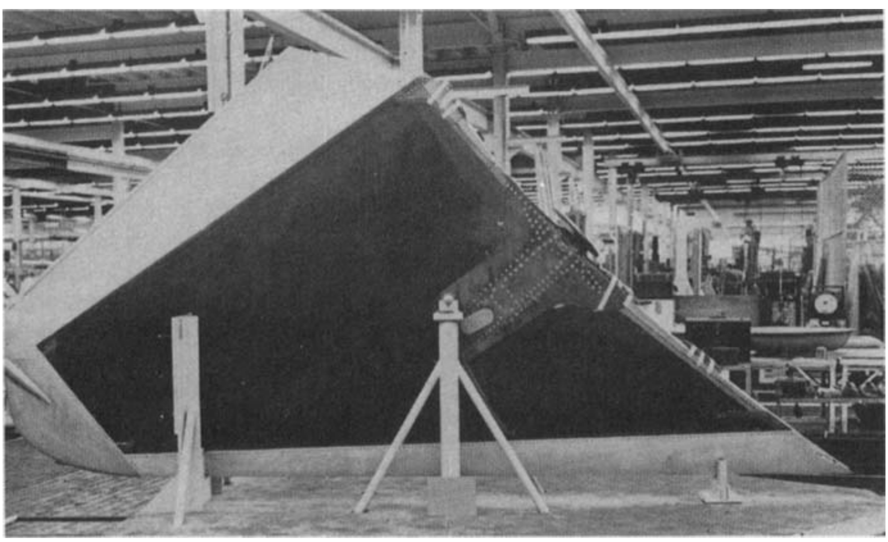
11.3 MILITARY AIRCRAFT

(1) F-4 (U.S.A.) — Rudder manufactured by using boron filament-reinforced epoxy skins over full-depth aluminum honeycomb

(2) F-14A (U.S.A) — Uses boron/epoxy (B/EP) skins on the horizontal stabilizer box, as shown in Fig. 11.3.1, a full depth honeycomb core and internal rib and spar structure. The stabilizer was designed so that no mechanical fasteners penetrate the boron skins



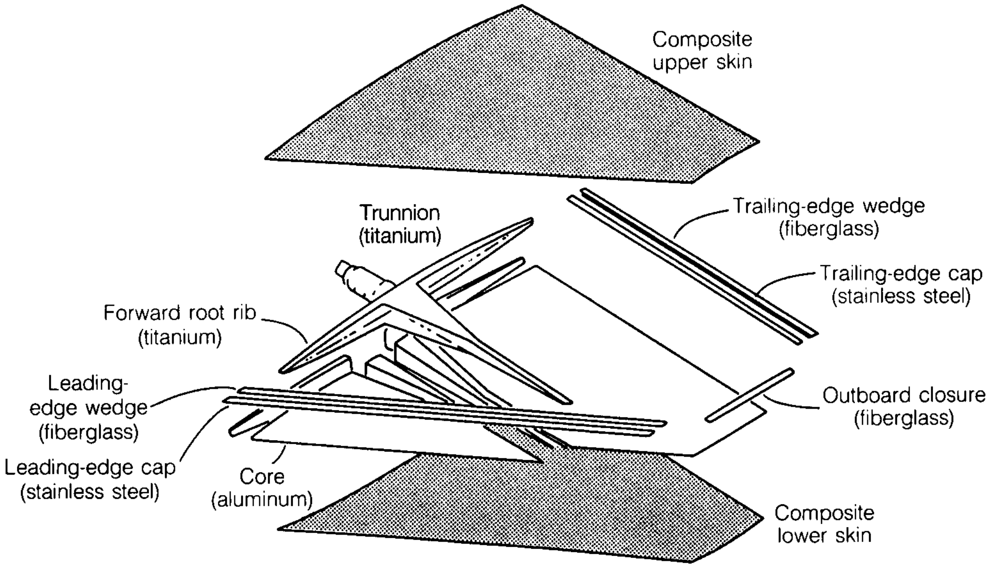
(a) Stabilizer details (full depth honeycomb core except between intercostals)

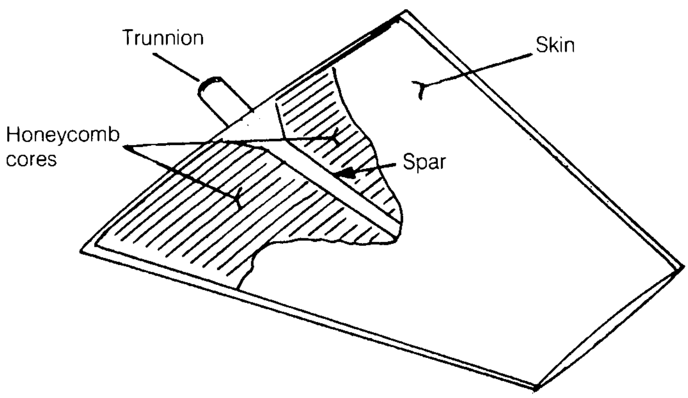


(b) Composite tail

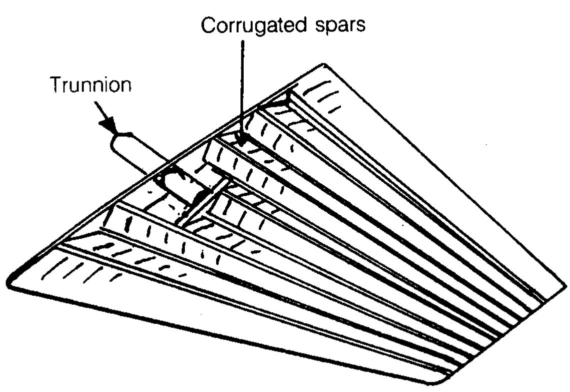
By courtesy of Grumman Corp.

Fig. 11.3.1 F-14A Boron/Epoxy Composite Horizontal Tail





(a) Full depth honeycomb (early design)



(b) Corrugated spar (aluminum) with composite skin (later design)

Fig. 11.3.2 F-16 Composite Horizontal Stabilizer

(3) F-15 (U.S.A) — Vertical fin, rudder and horizontal stabilizer skins and speed brake use boron/epoxy (B/EP)

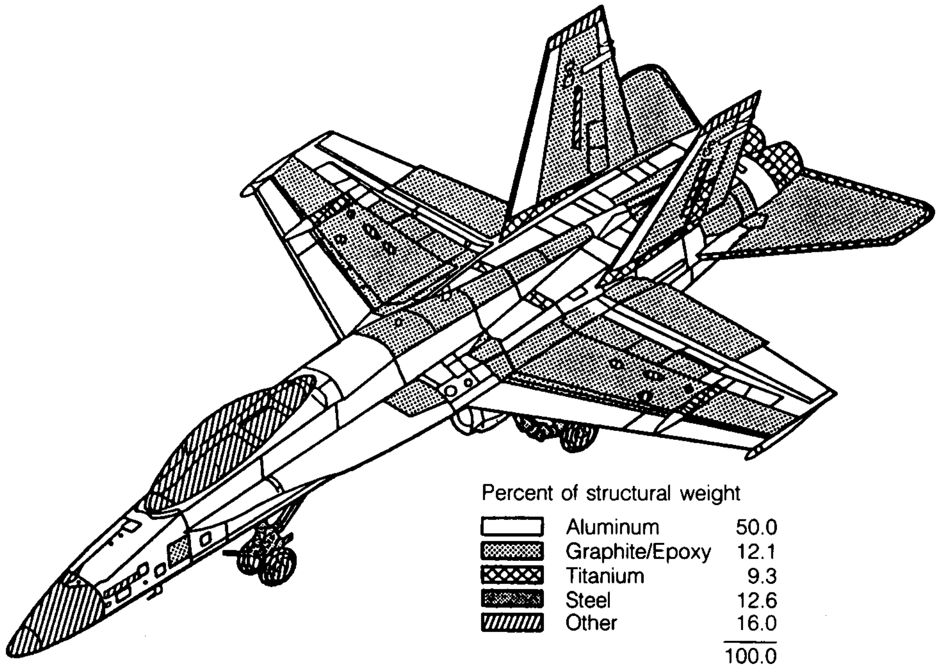
(4) F-16 (U.S.A) — Uses graphite/epoxy (GR/EP) skins on the vertical fin box, fin leading edge, rudder and horizontal tail. The fin leading edge, rudder and horizontal tails [see Fig. 11.3.2(a)] are all full depth aluminum honeycomb sandwich structures with graphite/epoxy facesheets.
Horizontal stabilizer construction [a later design shown in see Fig. 11.3.2(b)]:
- Graphite/epoxy (GR/EP) used on both upper and lower skins
- Full-depth corrugated aluminum truss core replaces honeycomb core
- Multiple layers of protection separate metal and composite surfaces
  - Anodized aluminum
  - Epoxy primer
  - Liquid scrim (epoxy and chopped glass)
  - Sealant
- Glass/epoxy cloth (over entire skin) is cocured to inner surface of graphite/epoxy skin
- Corrosion-resistant steel fasteners with sealant

(5) F-18 (U.S.A) — Graphite/epoxy (GR/EP) is used on wing skin, horizontal and vertical tails, control surfaces, speed brake, leading edges, and doors, and account for 12.1% of aircraft structural weight.(see Fig. 11.3.3)

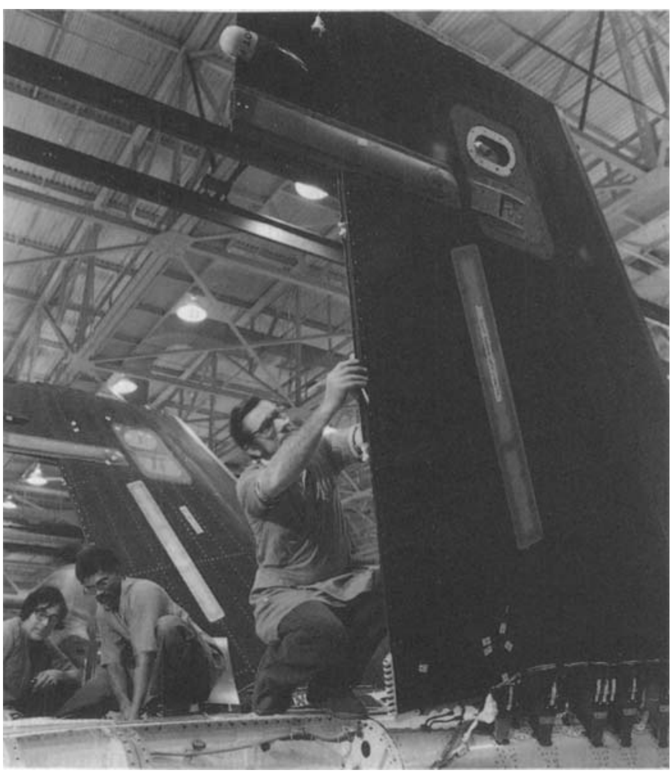
(6) AV-8B (U.S.A) — Fig. 11.3.4 shows graphite/epoxy (GR/EP) material is used in the wing box skins and substructure, forward fuselage, horizontal stabilizer, elevators, rudder, overwing fairing, engine bay door ailerons and flaps.
- One-piece solid laminate panel for both upper and lower wing skin (tip to tip), as shown in Fig. 11.3.5
- One of the unique features of the AV-8B aircraft is extensively use of sine-wave spars consisting of roll-formed webs with the caps attached, as shown in Fig. 11.3.6.
- The horizontal stabilizer box (tip to tip) is a carbon/epoxy multi-spar design with a one piece upper cover attached with special titanium blind fasteners, as shown in Fig. 11.3.7. The upper and lower covers are made from carbon/epoxy tape and the integral spars are channels made from woven cloth

(7) X-29A (U.S.A) — The X-29A, as shown in Fig. 1.1.8, uses graphite/epoxy (GR/EP) on the forward swept wing skins (156 plies at the thickest portion) and canard. Here "aeroelastic tailoring" is used to resist the characteristic nose-up twisting under load of the forward swept wing and to maintain the wing's structural, built-in twist throughout the flight envelope, delaying divergence.

(8) B-1B (U.S.A):
- Boron/epoxy was used to reinforce the dorsal longeron
- Weapons bay door, aft equipment bay doors and flaps are fabricated from full-depth aluminum honeycomb bonded with carbon/epoxy facesheets, except for the lower surface of the weapons bay door on which an Aramid fiber/phenolic outer layer provides damage resistance



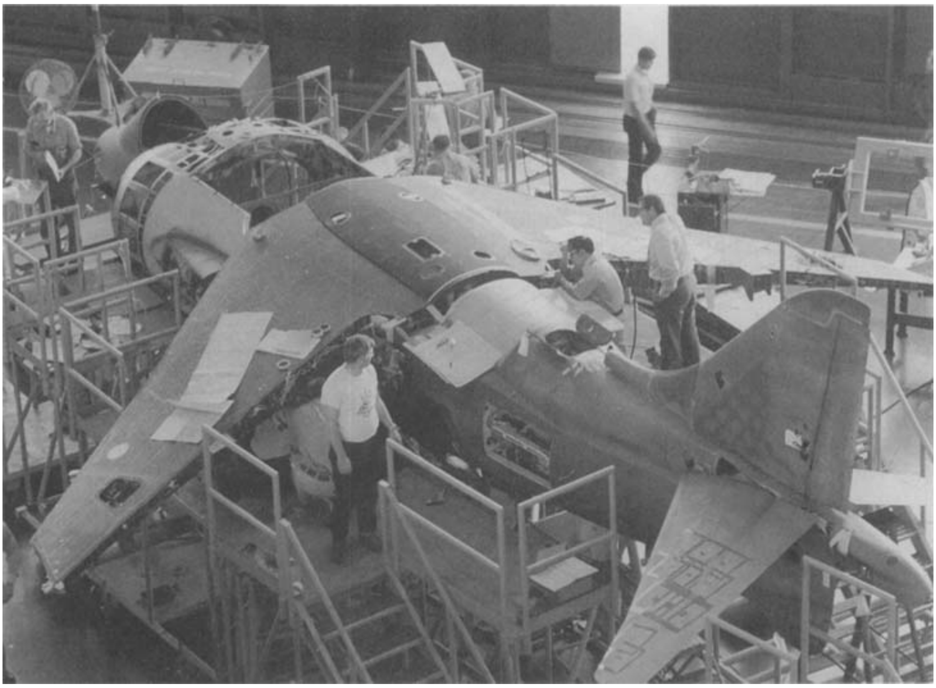
(a) Composite applications



By courtesy Northrop Corp.

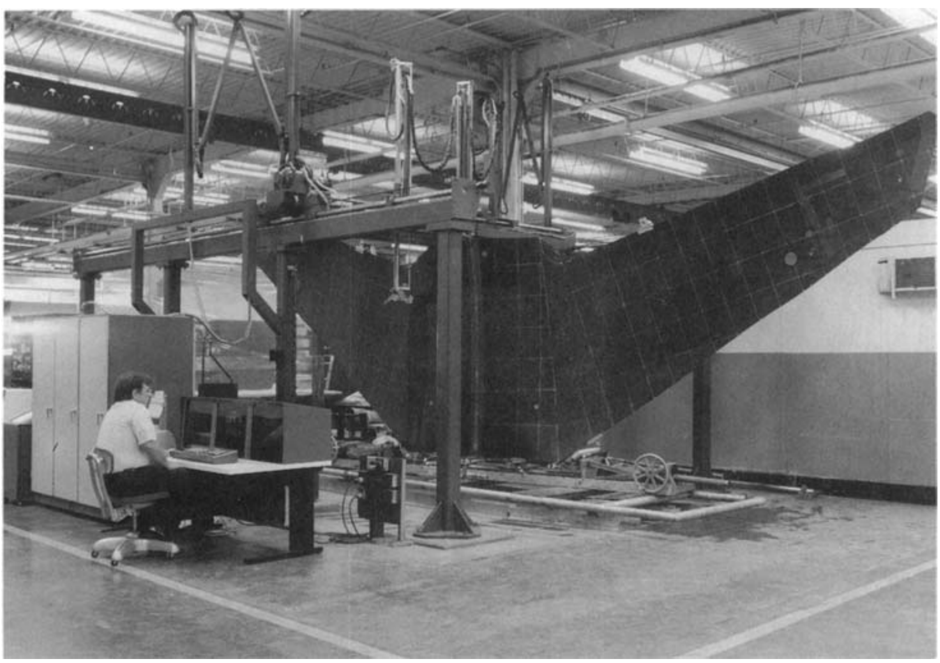
(b) Graphite/epoxy composite vertical tail

Fig. 11.3.3 F-18 Composite Application



By courtesy of McDonnell-Douglas Corp.

Fig. 11.3.4 Composites Are Used Extensively On The AV-8B (26.3%)



By courtesy of McDonnell-Douglas Corp.

Fig. 11.3.5 AV-8B Lower Wing Skin Which is 28ft (8.53m) Across Wing Tips And 121 ft^2 (11.24m^2) in Area

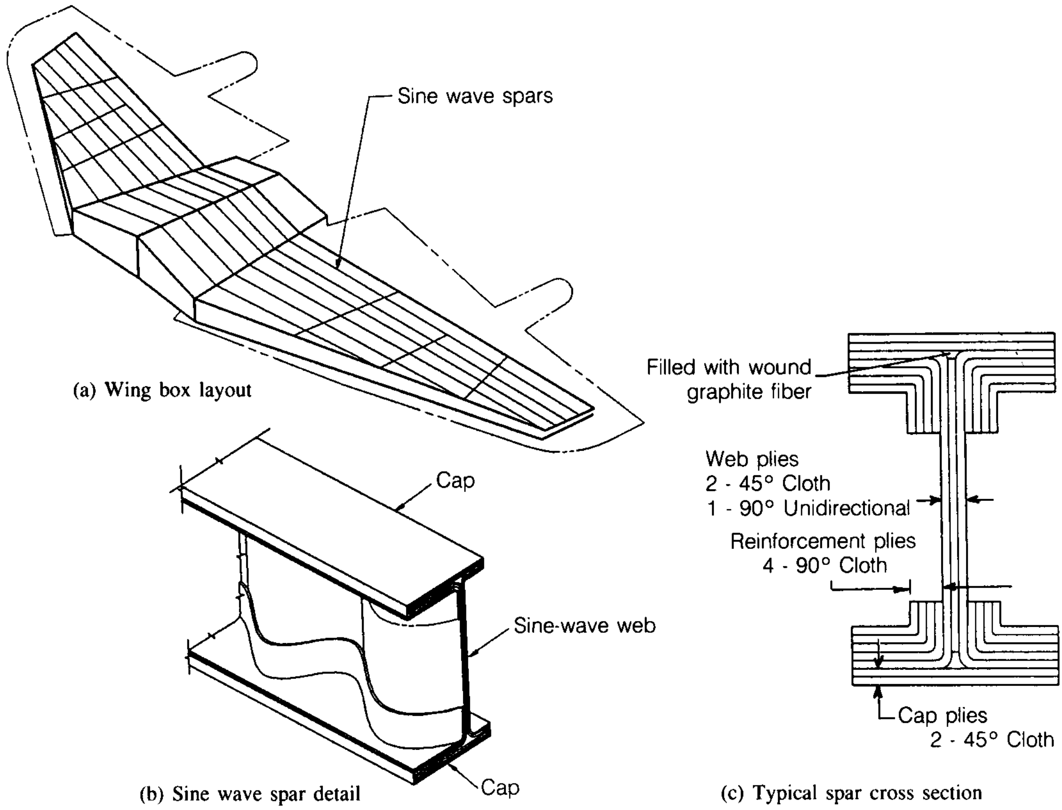


Fig. 11.3.6 AV-8B Composite Sine Wave Spar

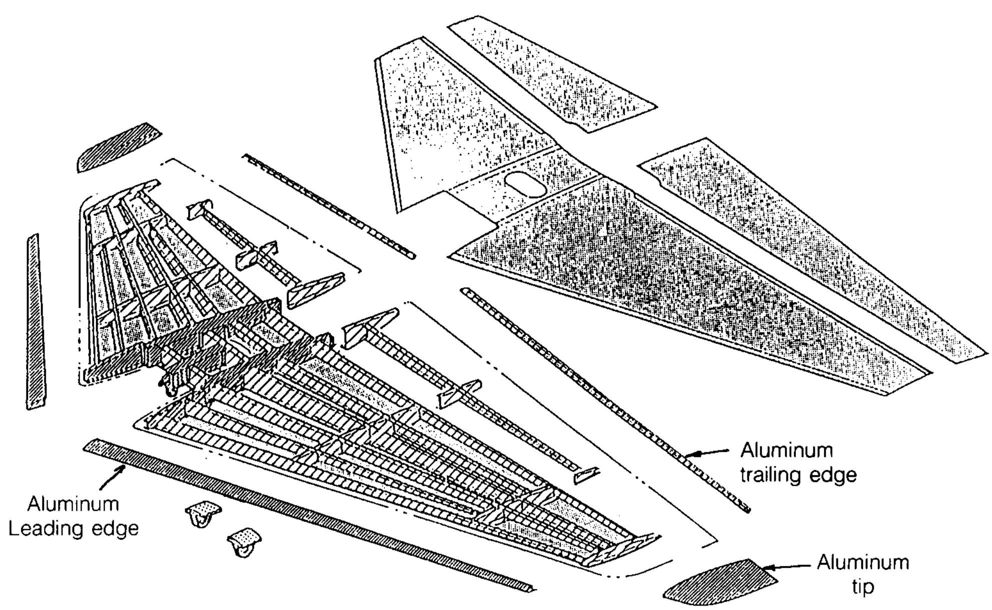


Fig. 11.3.7 AV-8B Composite Horizontal Tail

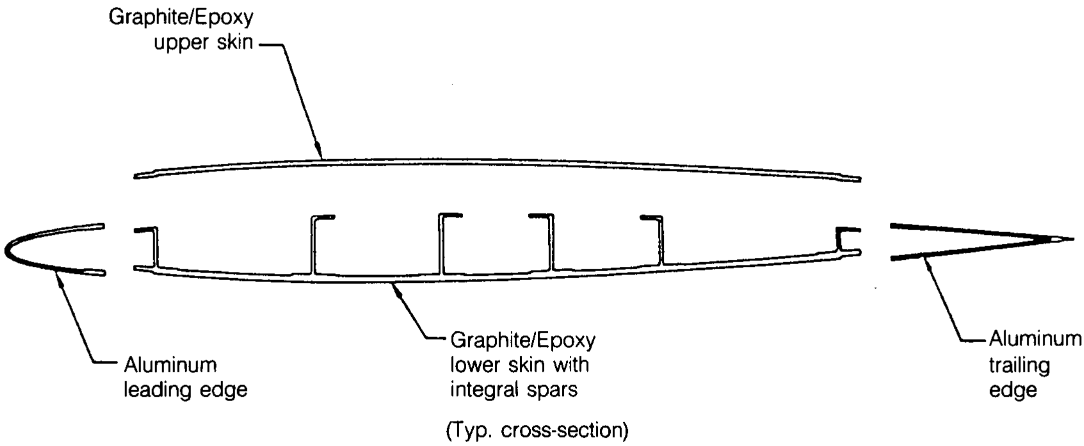


Fig. 11.3.7 AV-8B Composite Horigontal Tail (cont'd)

- Experimental composite horizontal and vertical stabilizers have been built for the B-1 bomber. Fig. 11.3.8 shows a vertical stabilizer constructed in a combination of graphite/epoxy (GR/EP) and boron/epoxy (B/EP) materials.

(9) V-22 V/STOL (U. S. A.):
- The wing consists of integrally stiffened composite laminate cover panels, with composite ribs and spars
- Fuselage is made of composite material which is 50% of its structural weight
- Wing leading and trailing edges are also made of composite materials
- Empennage is made of CFRP employing sine-wave spars

(10) A-6 Wing (U. S. A):
- The carbon/epoxy composite replacement the A-6 wing box is shown in Fig. 11.3.9
- The cover skins are fabricated as a one-piece panel using unidirectional tape
- Composite intermediate ribs and spars
- Titanium front and rear spars, and inboard and outboard tank end ribs.
- All parts are assembled by mechanical fastening

(11) Alpha Jet (France and Germany) — Dornier uses composites on trimmable stabilizer box (see Fig. 11.3.10.) for the Alpha Jet.

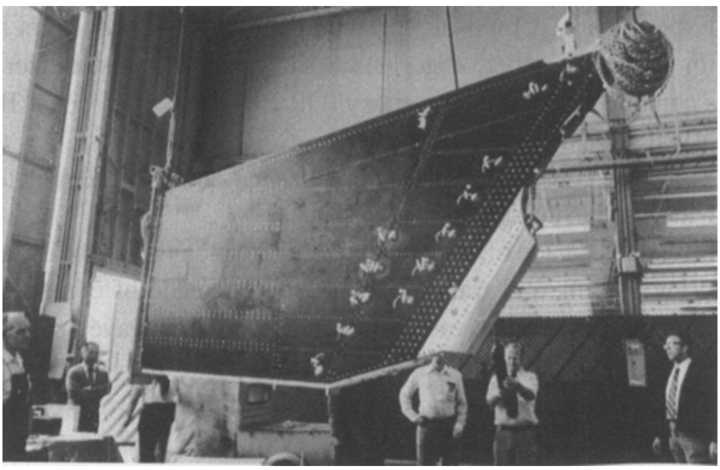
(12) Jaguar (British) — Engine bay doors and Wing box structures. The wing box, as shown in Fig. 11.3.11, is a bolted multi-spar and rib configuration:
- The front and rear spars are composite channel sections
- Five intermediate spars are sine wave composite beams
- The wing skins are predominantly $\pm 45°$ layup, which has low notch sensitivity (soft skin design approach)
- Sandwiched between the wing skins and the spar flanges are booms of predominantly $0°$, layup
- Wing bending is reacted mainly by the booms while shear and torsion are carried by the spar webs and the wing skins

(13) Tornado (British BAe and German MBB) — Composite taileron construction:
- The main box has composite skins with full-depth aluminum honeycomb

- The skin thickness varies from 0.128 inches (3.25 mm), at the root to 0.03 inches (0.75 mm) at the tip

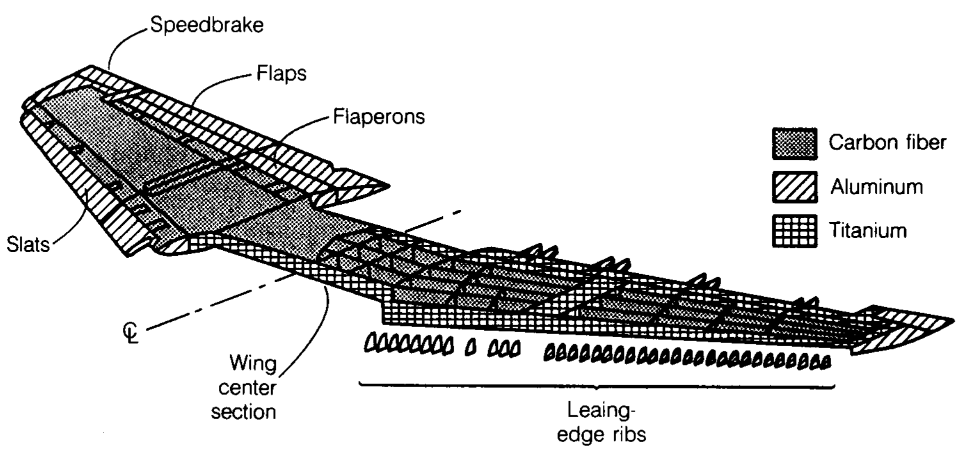
(14) EAP (Experimental Aircraft Program) — Wing, forward fuselage, canard and vertical tail will be built from composite materials. Fig. 11.3.12 shows an EAP wing design layout:
- Two cell fuel tank with notch spars
- No planks in skin (one-piece skin)
- All 0°, ±45° and 90° plies carbon/epoxy material
- Lower skin has co-bonded spars
- Upper skin is mechanically fastened with permanent fasteners

The composite wing construction of the European Fighter Aircraft (EFA) follows the same approach design as that shown in Fig. 11.3.13.



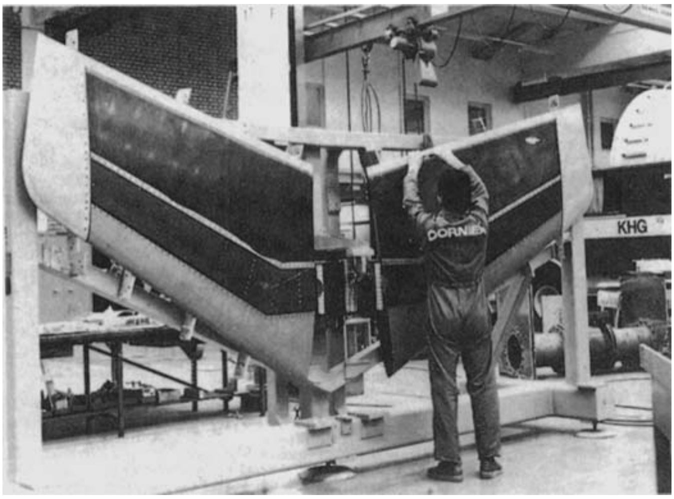
By courtesy of Rockwell International Corp.

Fig. 11.3.8 B-1 Composite Vertical Stabilizer



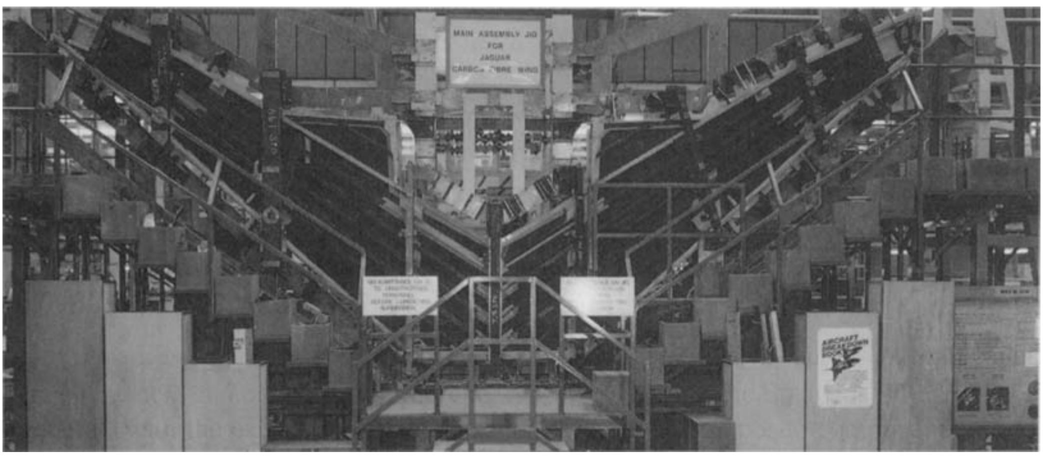
By courtesy of The Boeing Co.

Fig. 11.3.9 A-6 Composite Wing Box



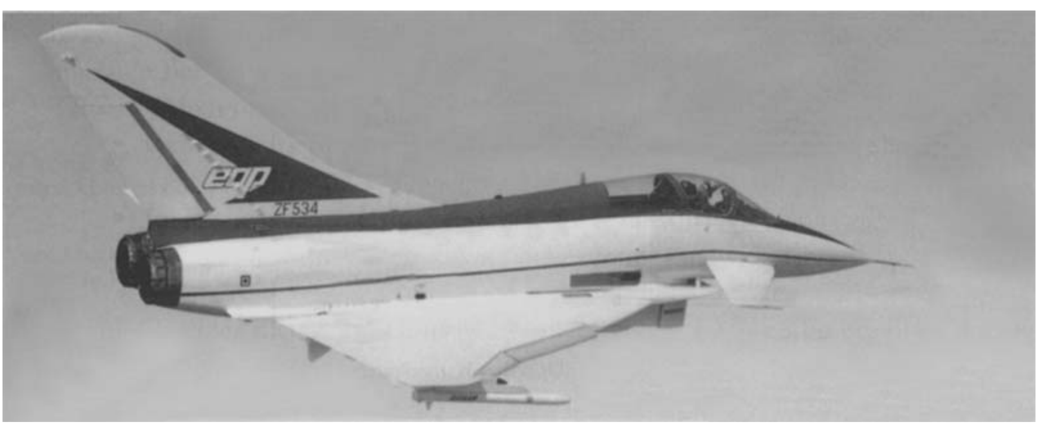
By courtesy of Dornier GMBH

Fig. 11.3.10 Alpha Jet Composite Tail



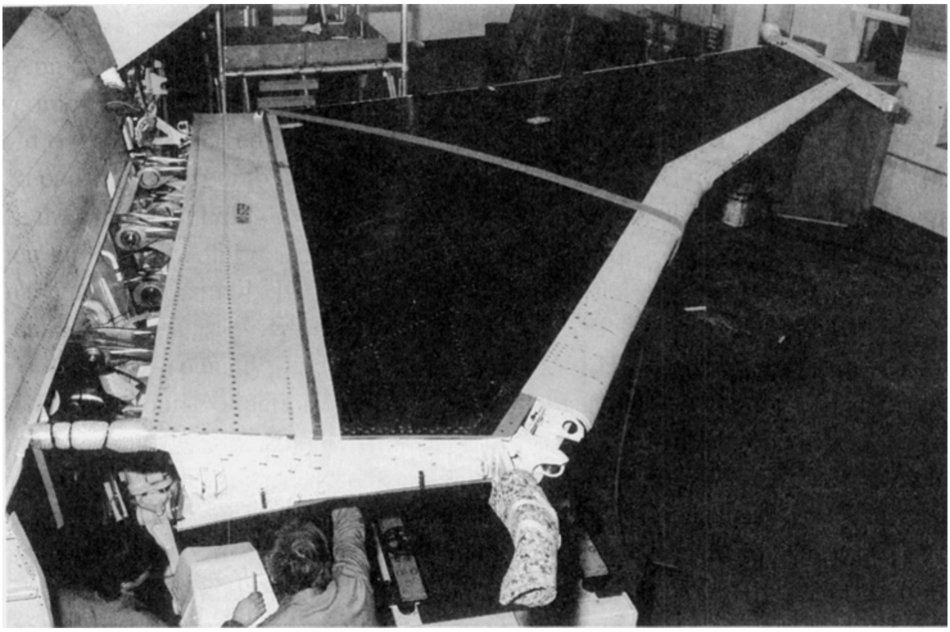
By courtesy of British Aerospace Military Aircraft Ltd.

Fig. 11.3.11 Jaguar Composite Wing Box



(a) EAP flight tests

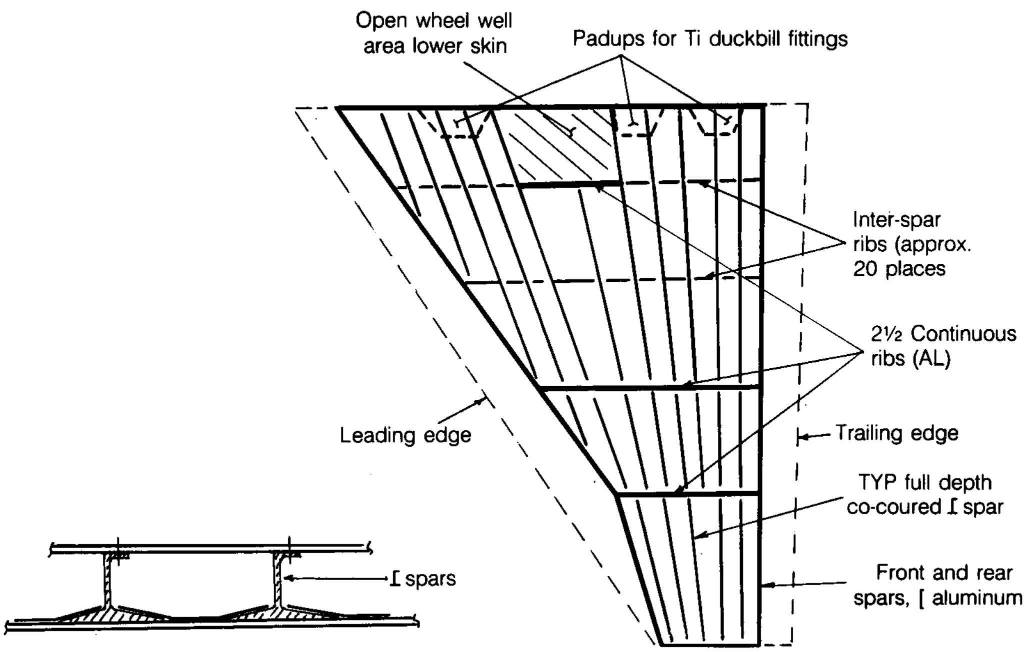
Fig. 11.3.12 British Experimental Aircraft Program (EAP) Demonstration Aircraft



(b) Wing box assembly uses Carbon/Epoxy; the multiple spars are cured and bonded to the lower wing surface in a single process

By courtesy of British Aerospace Ltd

Fig. 11.3.12 British Experimental Aircraft Program (EAP) Demonstration Aircraft (cont'd)

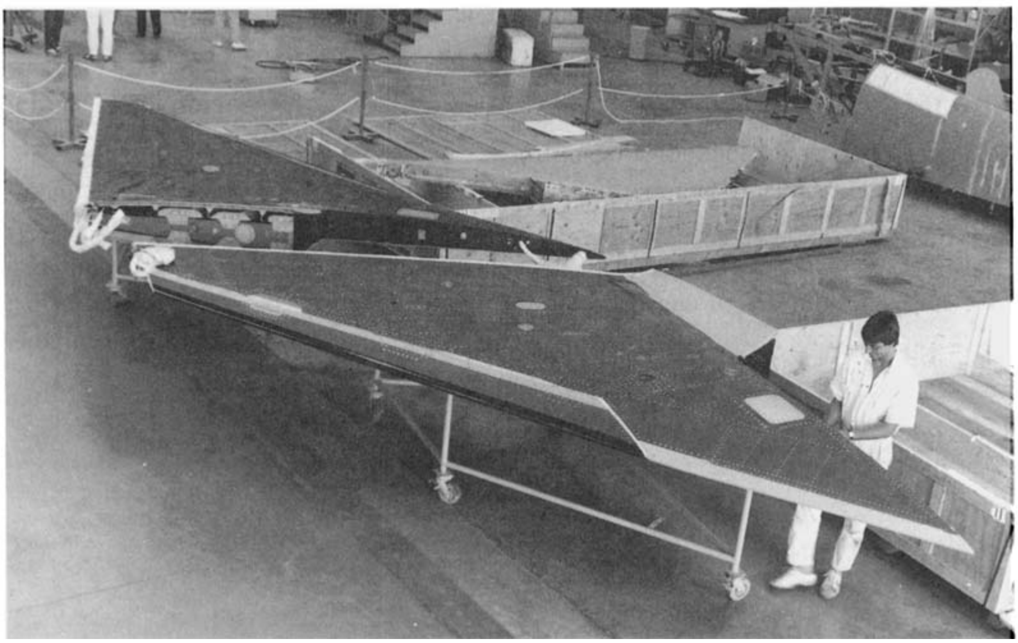


(Lower skin with co-cured spars)

(Upper skin, mechanically attached with permanent fasteners)

Fig. 11.3.13 European Fighter Aircraft (EFA) Wing Concept

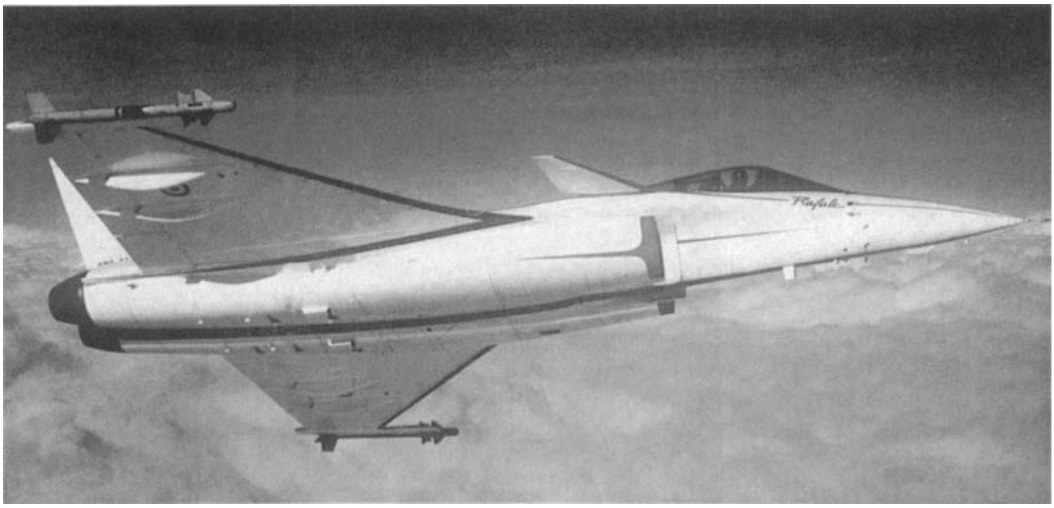
(a) Lavi aircraft



(b) Composite wing box (Fabricated by Grumman Corp.)

By courtesy of Israel Aircraft Industries Ltd.

Fig. 11.3.14 Lavi Composite Wing



By courtesy of Aerospatiale

Fig. 11.3.15 Rafale Aircraft

By courtesy of SAAB-SCANIA AB, SAAB aircraft division

Fig. 11.3.16 SAAB JAS 39 Gripen

(15) Lavi (Israel — Grumman Corp. built the composite wing) — Composite material is used on wing skins (see Fig. 11.3.14), vertical tail, moving foreplane, control surfaces, various doors and panels, and fuselage.

(16) Rafale (France) — Composite material is used on the wing, fuselage (50%), control surfaces, canard, landing gear doors and access doors (see Fig. 11.3.15)

(17) Gripen (Sweden) — Gripen design (see Fig. 11.3.16) uses composite material on the wing, vertical tail, canard, intake duct and landing gear doors.

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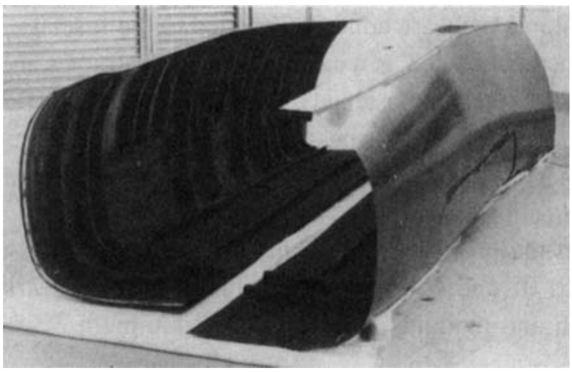
(18) MBB Fighter Fuselage Program (Germany) — Carbon material is used on the complete combat-aircraft forward fuselage (see Fig. 11.3.17) which is a two-piece fuselage shell featuring integrally cocured frames, stringers, and longerons.

(19) F-111 (U.S.A.):
- Boron doubler reinforces the overwing pivot joint structure
- Horizontal stabilizers (boron/epoxy skin over full-depth aluminum honeycomb) were fabricated for flight test.

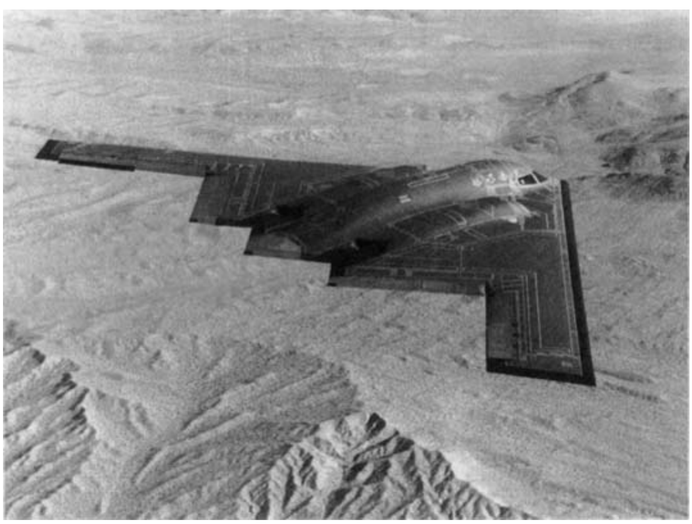
(20) B-2 (U.S.A.) — The B-2 (see Fig. 11.3.18) is the biggest all-composite airframe structure ever built

(21) ATF (U.S.A.) — There is extensive use composite materials (nearly 40%) on the Advanced Tactical Fighter (see Fig. 1.1.2).



By courtesy of MBB GMBH

Fig. 11.3.17 MBB Composite Forward Fuselage (Fighter)



By courtesy of Northrop Corp.

Fig. 11.3.18 All-composite Airframe B-2 Flying Wing Aircraft

Chapter 12.0

INNOVATIVE DESIGN APPROACHES

12.1 INTRODUCTION

Many current structural designs result in high manufacturing costs which limit preclude the application of advanced composites to other than subsidized hardware. Many of the unique problems of composite structural design have not been adequately addressed in either the current hardware programs or study efforts now underway. Early composite development programs utilized designs with structural configurations similar to those of metal counterparts. Hence, those designs embodied many of the undesirable characteristics of the past (e.g., excessive numbers of detail parts, complicated tooling and assembly techniques, etc.) with their only redeeming feature being lighter weight. Designs, materials, tooling concepts, and manufacturing methods are needed which take maximum advantage of relative low-cost processes such as filament (or tape) winding and resin transfer molding (RTM) techniques, and which can be single-stage cured at the required elevated temperature and pressure, without sacrificing structural integrity.

To obtain the utmost weight and cost savings potential of advanced composites in advanced aircraft structures, material forms and manufacturing methods must be taken into considerations during the development of innovative structural designs. This goal can be achieved by meeting the following objectives:

- Reduction of aircraft life cycles cost by advancing the state-of-the-art of composite innovative design concepts
- Paving the way for manufacturing and producibility methods that will find practical usage in airframe structures
- Incorporation of low cost manufacturing, tooling, assembly and repair methods

In order for a breakthrough to be made in composite structural technology, innovative composite design concepts are crucial. In general, innovative concepts can be categorized into two groups:

- Ideal or perfect design — impossible today (long term goal)
- Realistic or practical design — Possible but needs more work (near term goal)

Design Requirements

However, structural weight reduction and cost savings cannot be accomplished without input from the concurrent engineering teams (but not limited to these):

- Structural design
- Manufacturing and producibility

- Tooling
- Materials and processes
- Maintainability and repairability
- Stress engineering

Also, emphasis on modular monolithic types of structure, with reduced part counts, must be balanced against more complicated or exotic tooling (a cost consideration), minimization of cutouts and holes, and the use of cocured or co-consolidated composite structures. Innovative structural concepts should feature novel designs that not only exploit the unique characteristics of composite materials but also utilize structural mechanics to validate the concepts.

Rules of Innovative Design Approaches

The following concepts are basic components of innovative composite designs:

- Fiber orientation concept
- Modular concept
- Monolithic (integral) concept

The innovative composite design concepts discussed here are only a few of the many possible and all the concepts shown in this chapter are concepts only and may need more detailed work prior to practical application. Bear in mind that the philosophy of composite design should be completely different than that of conventional metal design which consists of many parts assembled together with numerous mechanical fasteners. In composite structures, elimination or most of all fasteners is critical not only to increase structural efficiency (eliminating composite material notched effect) and save structural weight, but also to reduce assembly cost. These innovative concepts should be considered a guide to composite engineers for the exploitation of true composite structural design and final achievement of the goal of more than 50% weight savings and more than 25% cost savings.

Incorporating manufacturing input during the creation of innovative concepts is strongly recommended to confine the design concept within reasonable costs as well as to insure technology availability within the near future. Another important aspect in developing new designs is to convince composite engineers to accept innovative concepts (e.g., by conducting structural tests which support the concept) and minimizing or eliminating structural conservatism, such as the widespread use of mechanical fasteners.

Patent Protection

It is obvious that composite design is not simply fabricating metal designs out of a new lighter material i.e., composites. Since the design of composite structures is a leading edge technology, many new concepts will be developed for weight reduction and cost-effective manufacturing processes. It is important to protect new ideas by filing by a patent as soon as possible so that priority rights to the invention may be secured.

12.2 FIBER ORIENTATION CONCEPT

In composite laminate structures, the fiber is the primary load-carrying material and should be used efficiently to avoid fiber interruption or termination. The fiber orientation concept accomplishes this by use of the following:

- Efficient usage of directional fibers, i.e., 0°, 45°, −45°, 90°, etc.
- Convergent and divergent concept
- Straight-through concept
- Wrap-around concept

(1) Efficient usage of directional fibers:

Since fibers are the primary load-carrying materials, it is extremely important in composite design that the fibers, and therefore the load path, are continuous rather than being disrupted by fiber termination at mechanically fastened spliced joints. Splice joints cause structural deficiency, i.e., eccentricity, notched effect (reducing laminate strength), fuel sealing problems, etc. Fig. 12.2.1 and Fig. 12.2.2 show examples which illustrate this concept.

(2) Convergent and divergent concept:

The convergent concept involves converging all or part of the fibers of a laminate panel into one or several fittings, lugs, etc. which carry a very highly concentrated load. The fighter wing root fittings and lugs, shown in Fig. 12.2.1 and Fig. 12.2.2, illustrate the convergent concept.

In contrast, the divergent concept involves diverging or spreading a concentrated load evenly through built-in or integral trunnions or fittings in a panel to moderate the local notched effect or stress concentration. The landing gear trunnion mounted bulkhead panel, shown in Fig. 12.2.3, is a typical case; it uses the fiber-disc or fiber-sector to spread the trunnion concentrated load into the bulkhead panel.

(3) Straight-through (Geodesic panel) concept

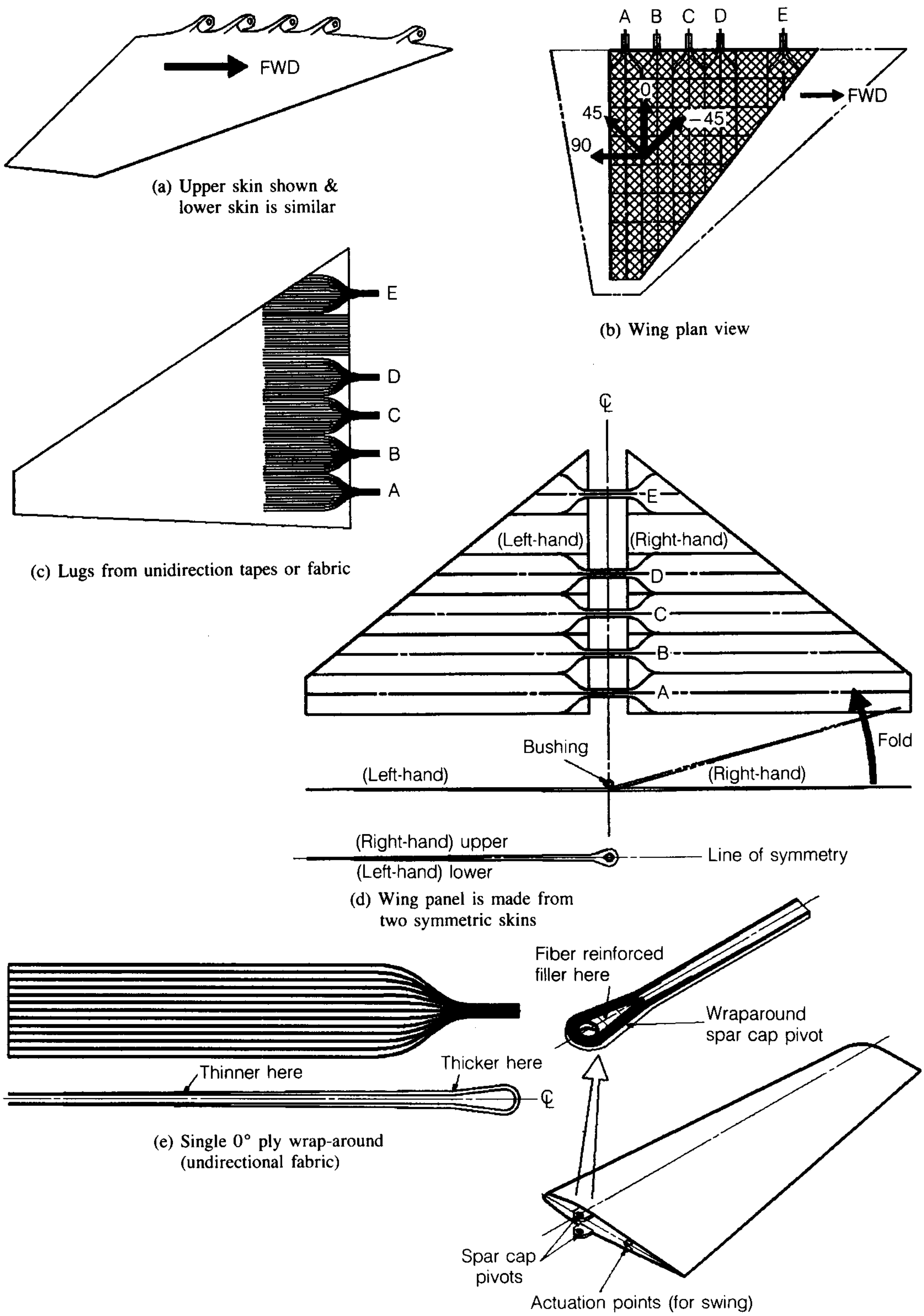
From a structural standpoint, the fibers in composite geodesic structural grid (orthogrid or isogrid) panels should be kept as straight as possible to maintain mechanical strength. An orthogrid panel (see Fig. 12.2.4) and similarly, the isogrid panel (see Fig. 12.2.5) is constructed by using longitudinal and transverse grid walls which intersect orthogonally to form a grid system that provides strength and stiffness to the panel, allowing both axial and lateral loads.

The bar-grid panel design concept, shown in Fig. 12.2.6, consists of alternating layers of consolidated composite material bars (round, square, rectangular, etc.) and syntactic material to form the grids.

(4) Wrap-around concept:

The critical location on the bar-grid panel is where the grid terminates at the panel edge. Both Fig. 12.2.3 and Fig. 12.2.6 illustrate how the terminating bars can wrap around over the bars forming the grid at the edge of the panel.

Fig. 12.2.1(f) is another type of wrap-around concept which wraps fibers around a pre-located metal bushing and forms a wing panel lug or canard spar root lug able to carry concentrated loads.



(a) Upper skin shown & lower skin is similar

(b) Wing plan view

(c) Lugs from unidirection tapes or fabric

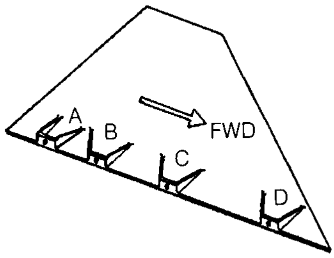
(d) Wing panel is made from two symmetric skins

(e) Single 0° ply wrap-around (undirectional fabric)

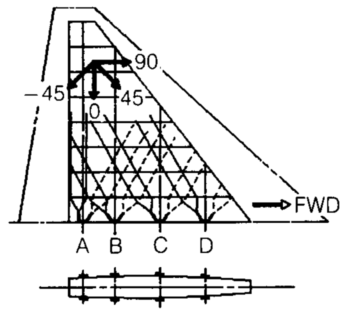
(f) Design example — starship canard spar root lug (also see Fig. 11.4.8)

By courtesy of Lockheed Aeronautical Systems Co.

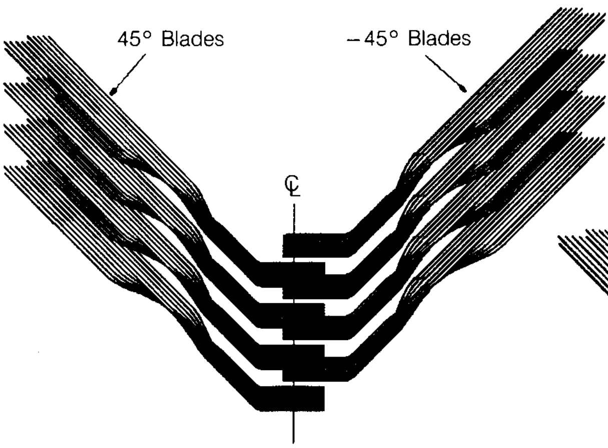
Fig. 12.2.1 Wing Root Integral Shear Lugs



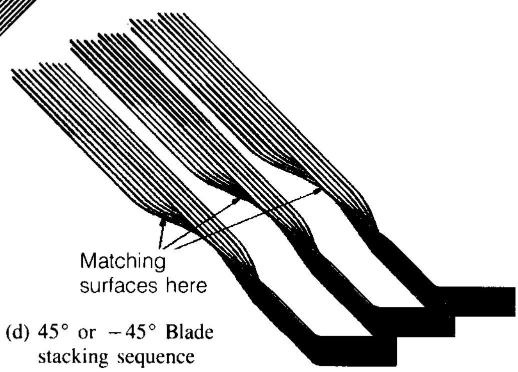
(a) Wing skin with tension fittings



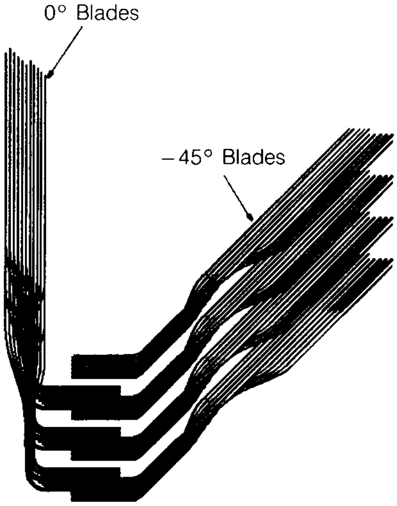
(b) Wing skin plan view



(c) For fitting B, C and D

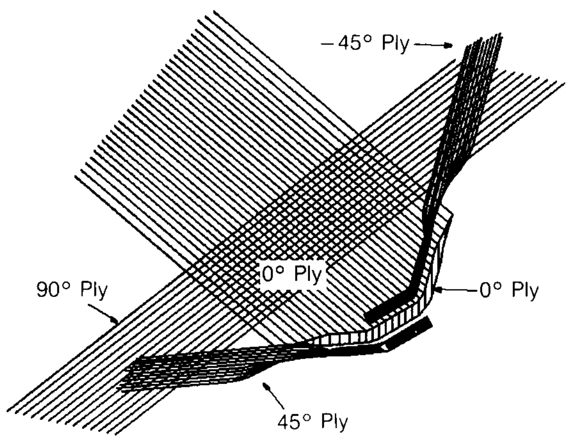


(d) 45° or −45° Blade stacking sequence



(For fitting A only)

(e) 0° And −45° Blades

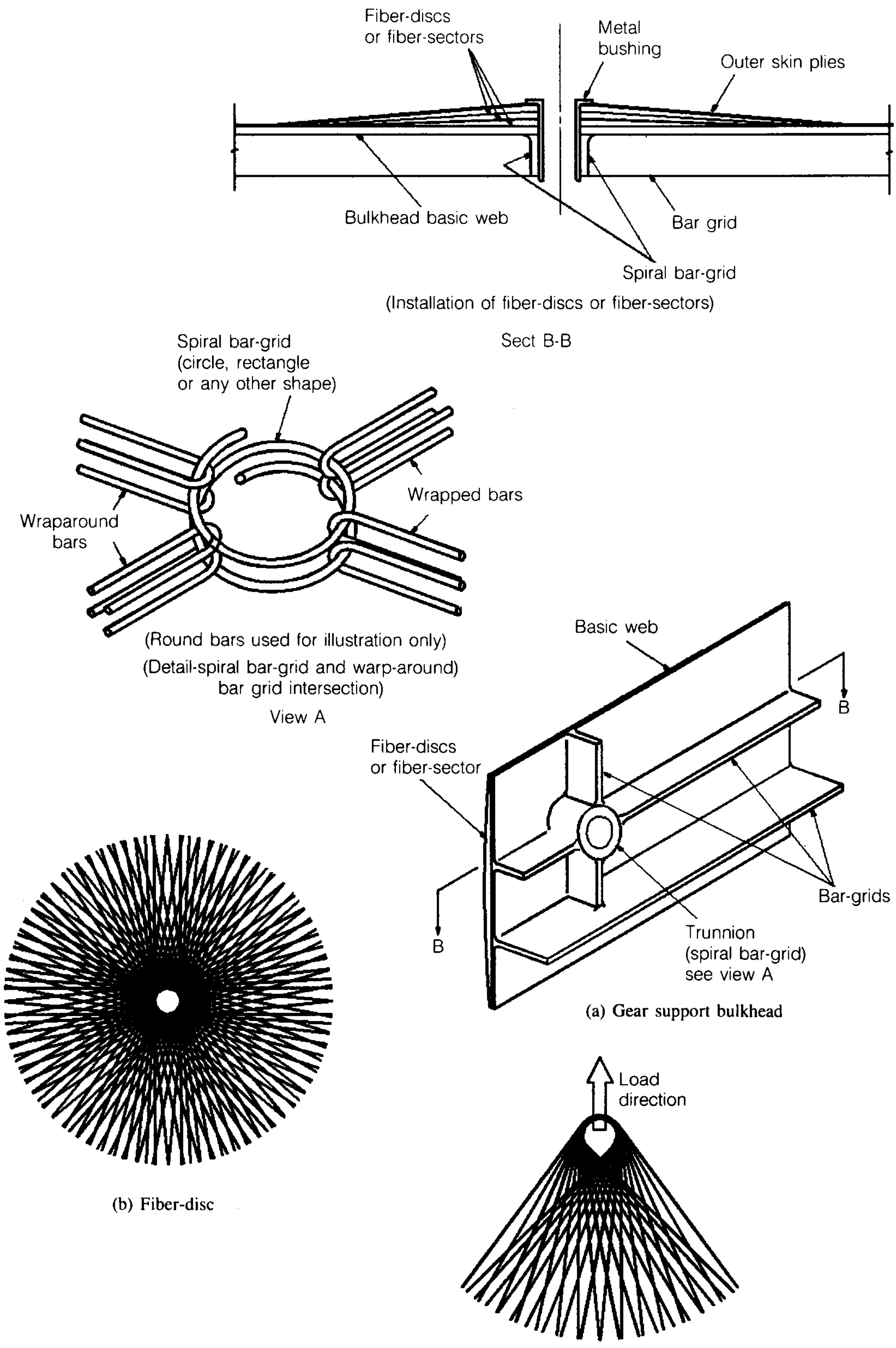


(f) Typical construction of tension fittings

(g) 0° Blade (half piece shown)

By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.2.2 Wing Root Integral Tension Fittings



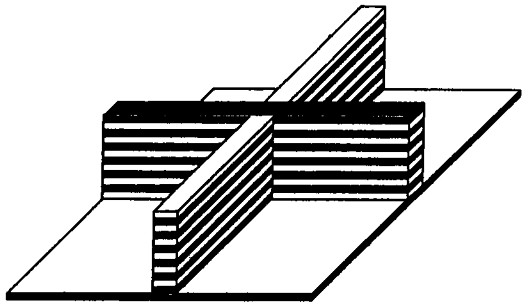
(a) Gear support bulkhead

(b) Fiber-disc

(c) Fiber-sector

By courtesy of Lockheed Aeronautical Systems Co.

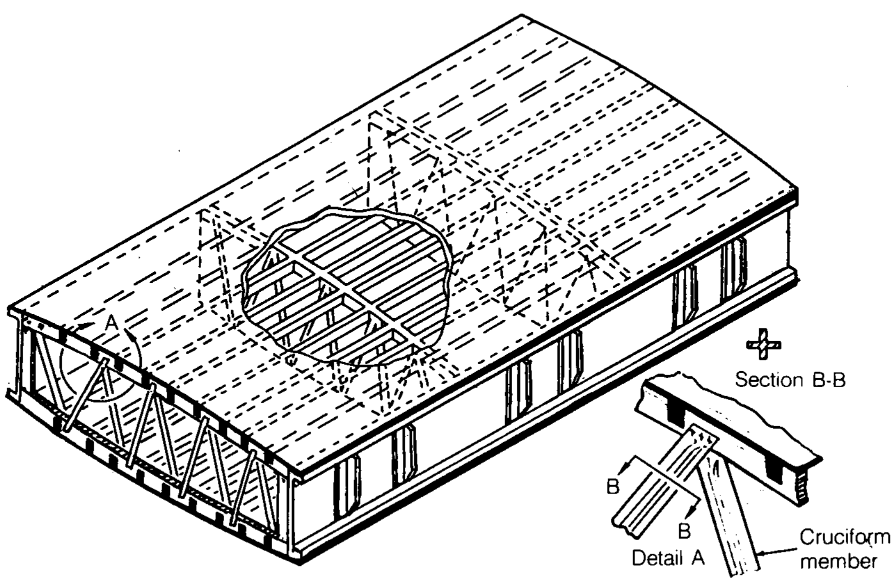
Fig. 12.2.3 Methods of Distributing Concentrated Load



(a) Detail of orthogrid (see Fig. 7.5.2)

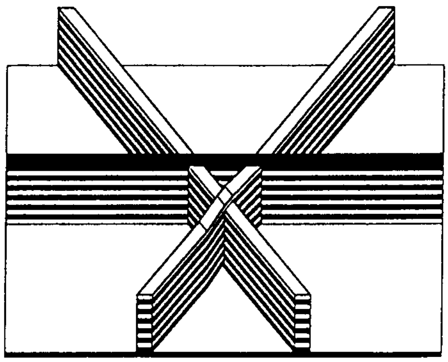
By courtesy of Lockheed Aeronautical Systems Co.

(b) Orthogrid panel and its tool

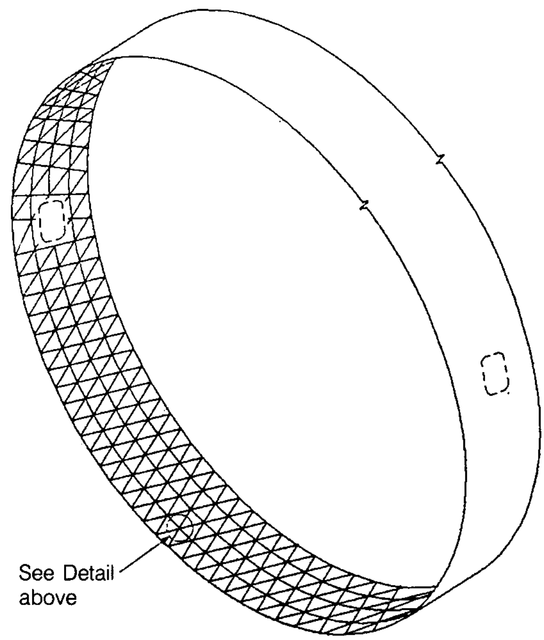


(c) Proposed orthogrid application (wing box)

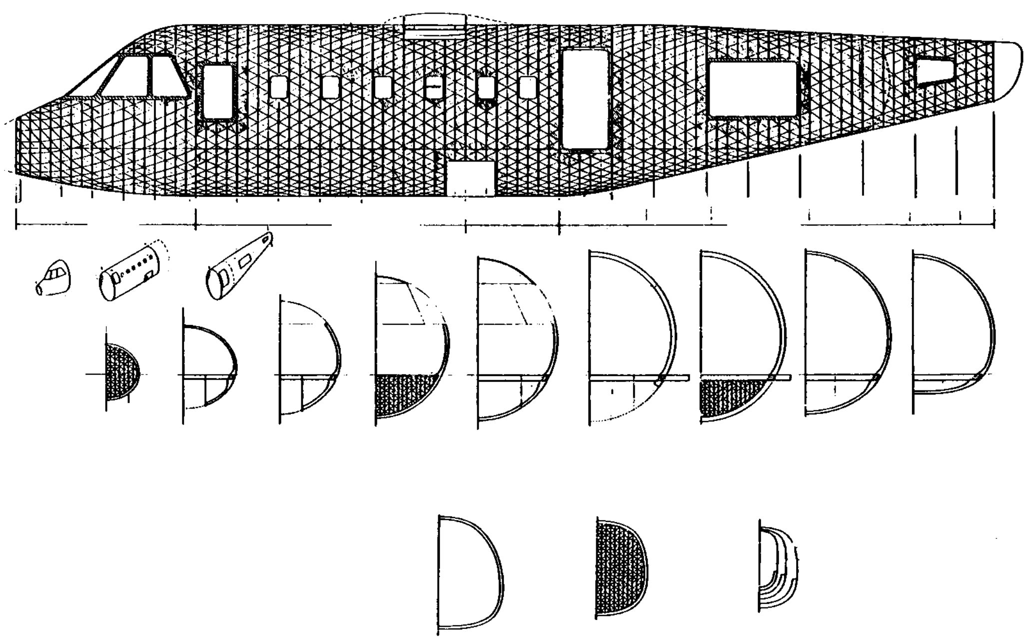
Fig. 12.2.4 Geodesic Structure (Orthogrid)



(a) Detail of isogrid (see Fig. 7.5.3)

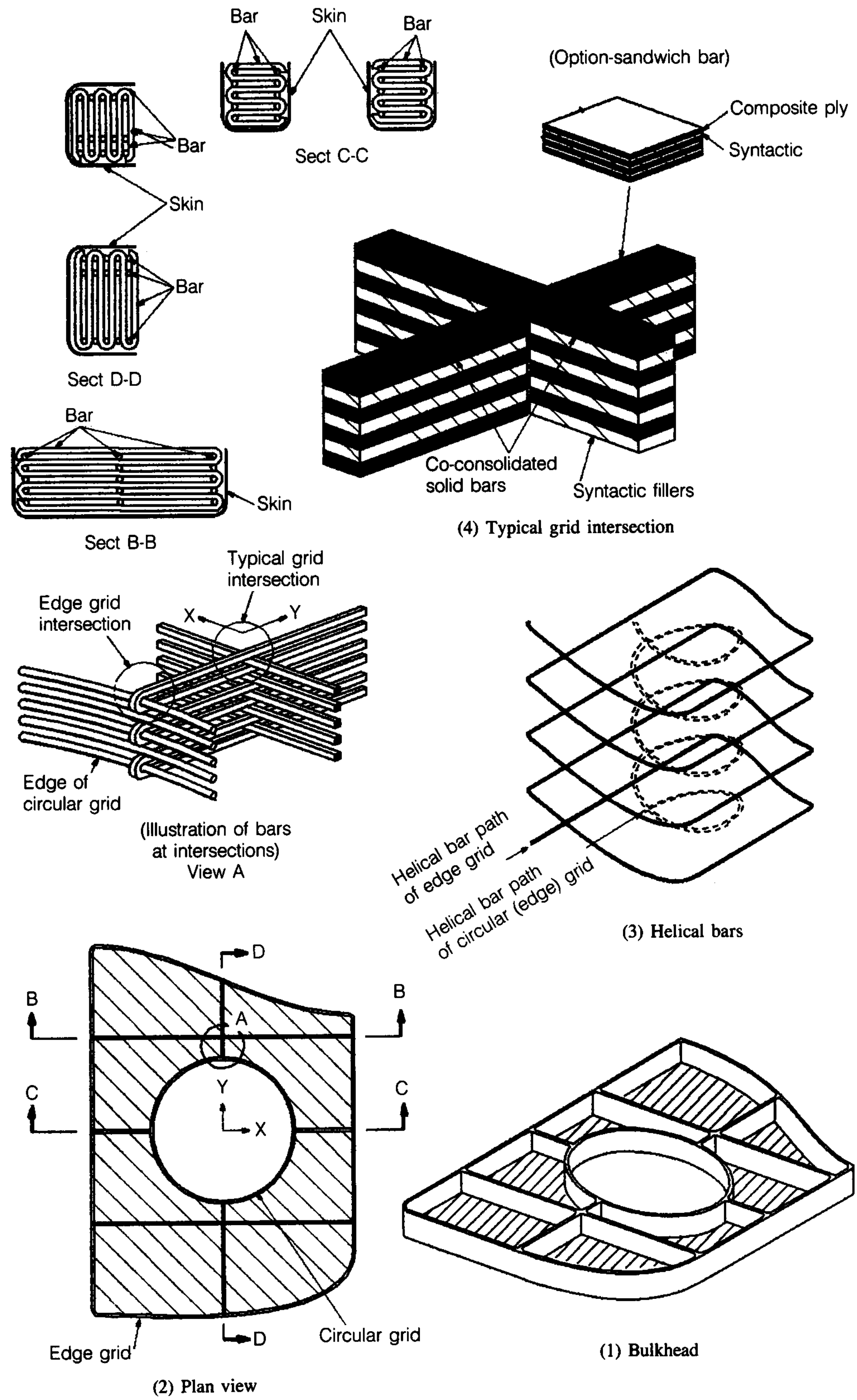


(b) Isogrid application (fuselage shell)



(c) Design example

Fig. 12.2.5 Geodesic Structure (Isogrid)



By courtesy of Lockheed Aeronautical Systems Co.

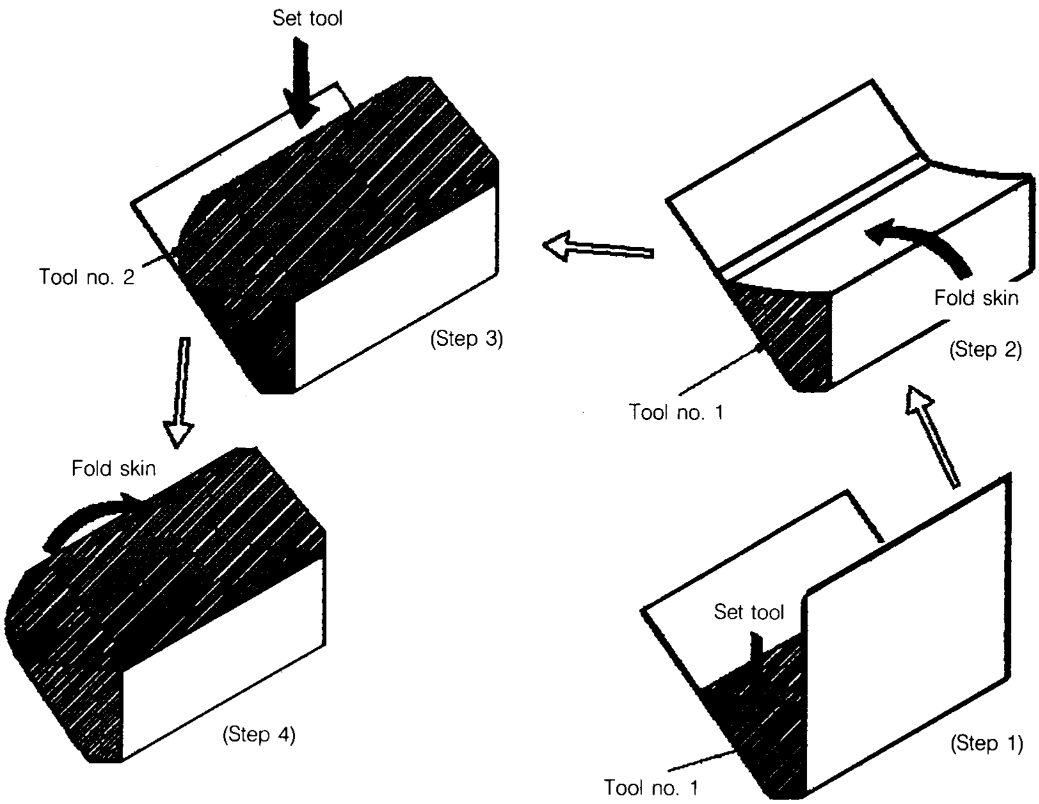
Fig. 12.2.6 Bar-Grid Panel

12.3 MODULAR CONCEPT

The modular concept involves an integral or monolithic type structure in which there are only a few major parts combined at final assembly rather than the conventional metal construction consisting of many small parts assembled with numerous mechanical fasteners. The modular concept may require complicated or special tooling to be accomplished successfully.

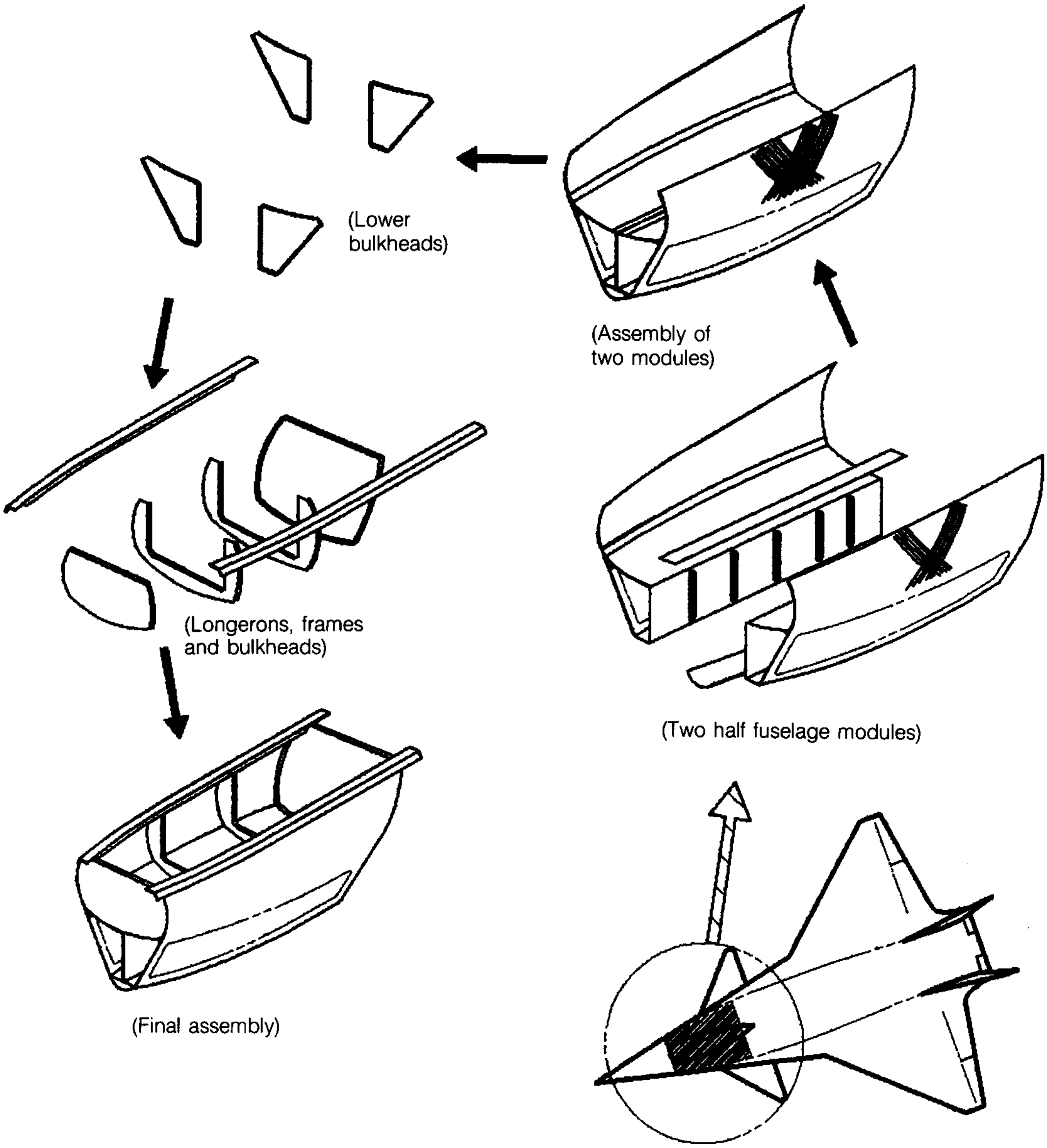
Fig. 12.3.1 illustrates a fighter fuselage modular design and Fig. 12.3.2 illustrates transport fuselage modular design; both concepts are assembled together by bonding (thermosets) or dual-resin bonding (thermoplastics) or another similar joining method. Design considerations for the modular concept are listed below:

- A simple component with little or no assembly
- Requires special mandrels and tooling
- Special cocure or co-consolidation technique needed
- No mechanical fasteners used for assembly
- May involve the use of sandwich, beaded, sine-wave or corrugated panels



(a) Tooling illustration (example only)

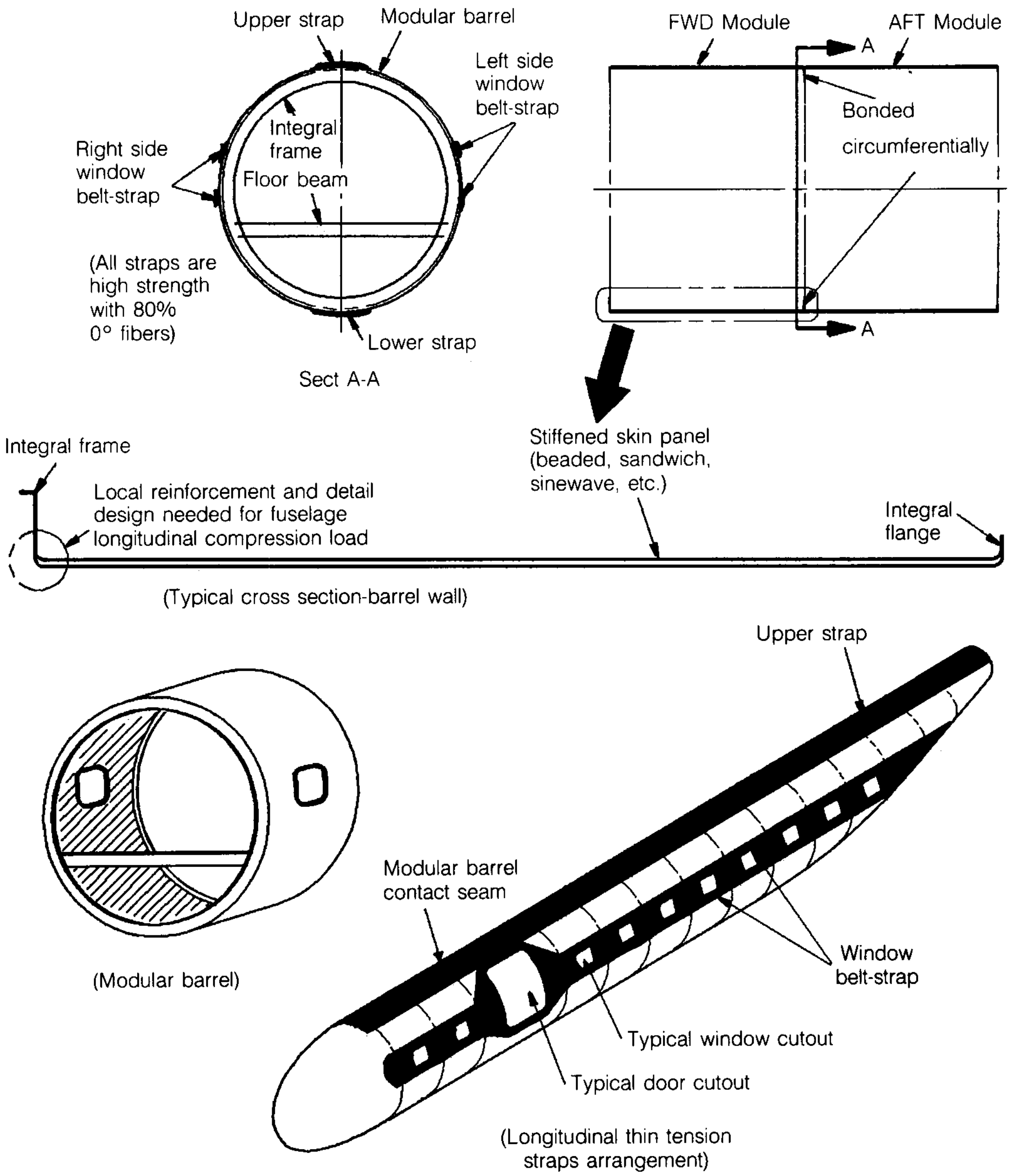
Fig. 12.3.1 Modular Design for Fighter Fuselage



(b) Assembly sequence

Fig. 12.3.1 Modular Design for Fighter Fuselage (cont'd)

To produce a modular concept design, innovative and out-of-the ordinary tooling is a must. As mentioned previously, with this type of structure reduced part count must be balanced against more costly tooling, but increased tooling cost will become only a fraction of the total cost once it is divided by the hundreds of units produced. This modular design concept has been applied to all-composite kit planes (see Section 11.4 in Chapter 11) for ease of assembly by amateur aviation builders.

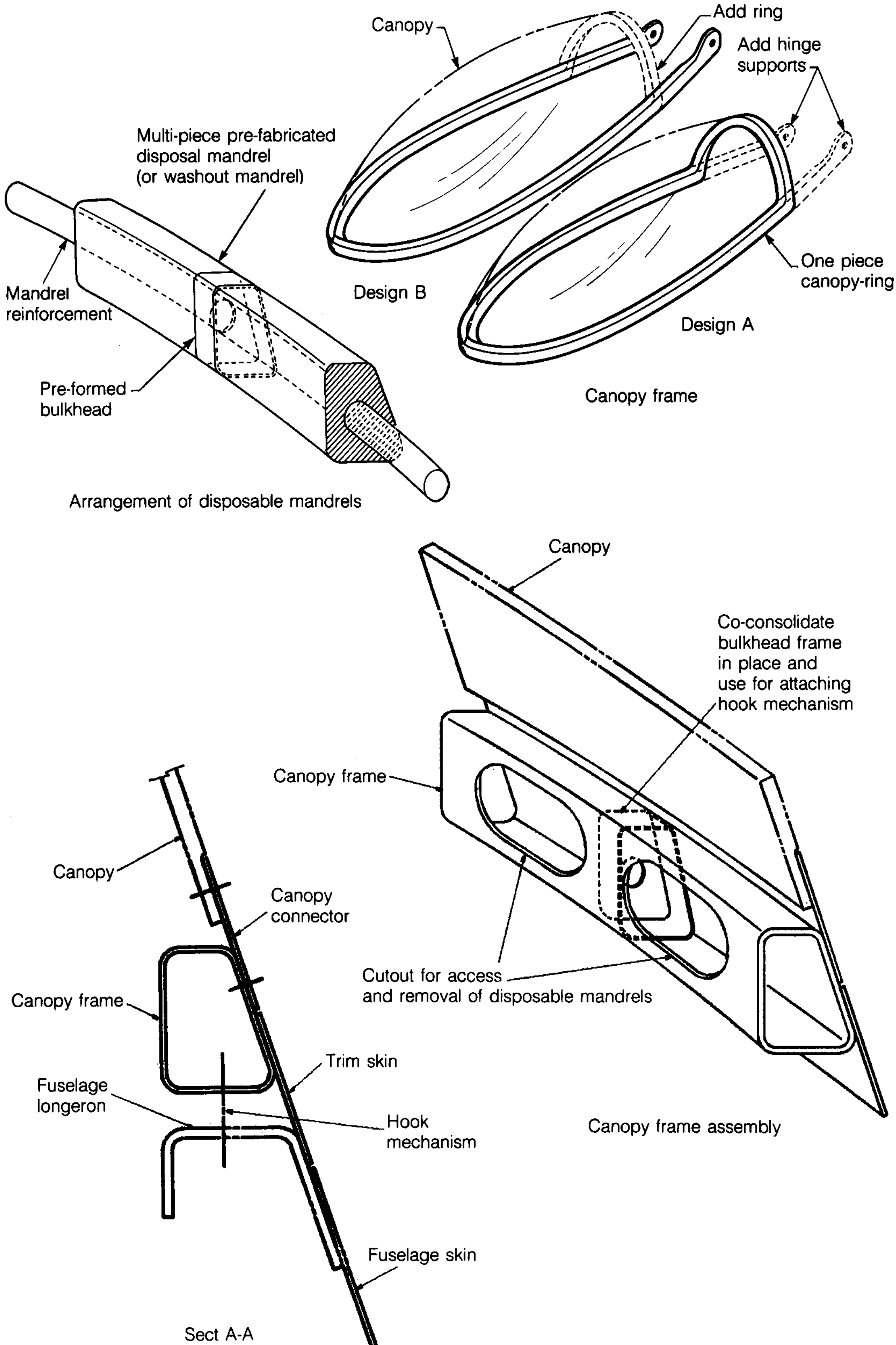


By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.3.2 Modular Barrel Fuselage

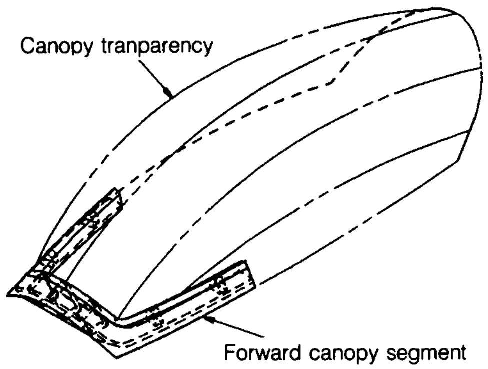
12.4 MONOLITHIC (INTEGRAL) CONCEPT

Fig. 12.4.1 shows a monolithic (one-piece) canopy frame which can be fabricated by low-cost filament winding using wash-out (disposable) mandrels (see Section 3.4 in Chapter 3) or other similar mandrels. This is a U-shaped frame with a constantly changing cross-section; all internal bulkheads, rear end hinges, etc. are built into the canopy frame by one cure operation. Both weight and cost savings are very substantial with this design concept.

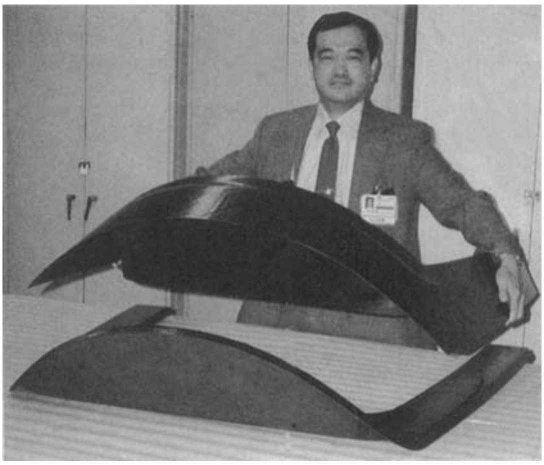


By courtesy of Lockheed Aeronautical Systems Co.

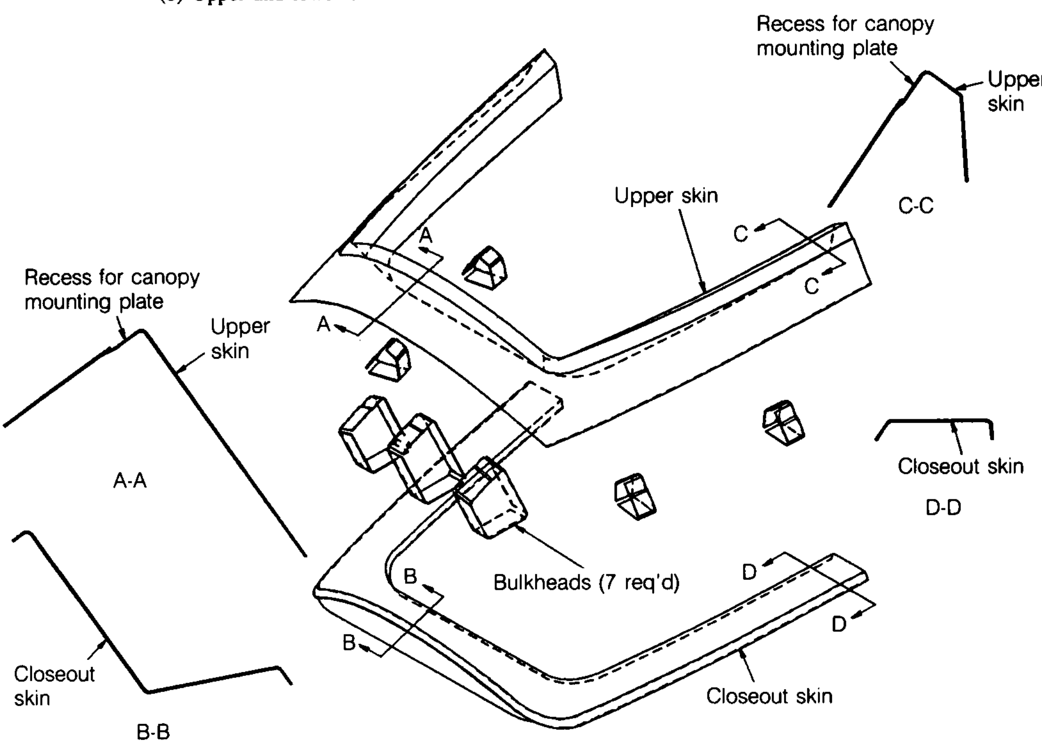
Fig. 12.4.1 One-Piece (Monolothic) Canopy Frame



(a) Forward composite canopy segment



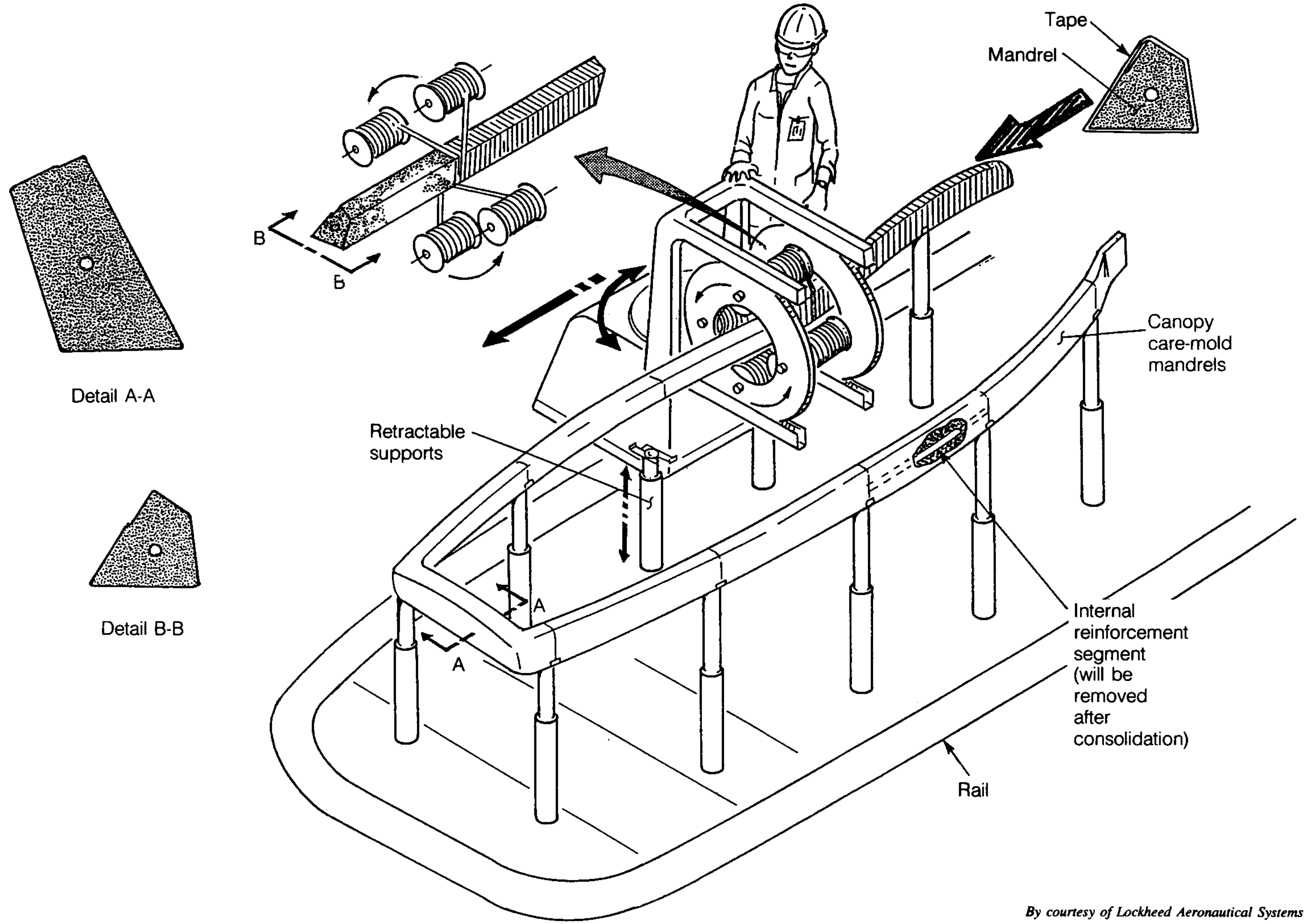
(b) Upper and lower skins



(c) Part breakdown of canopy segment

By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.4.2 Proof-of-the-Concept of Forward Composite Canopy Segment (Hand Layup Method)



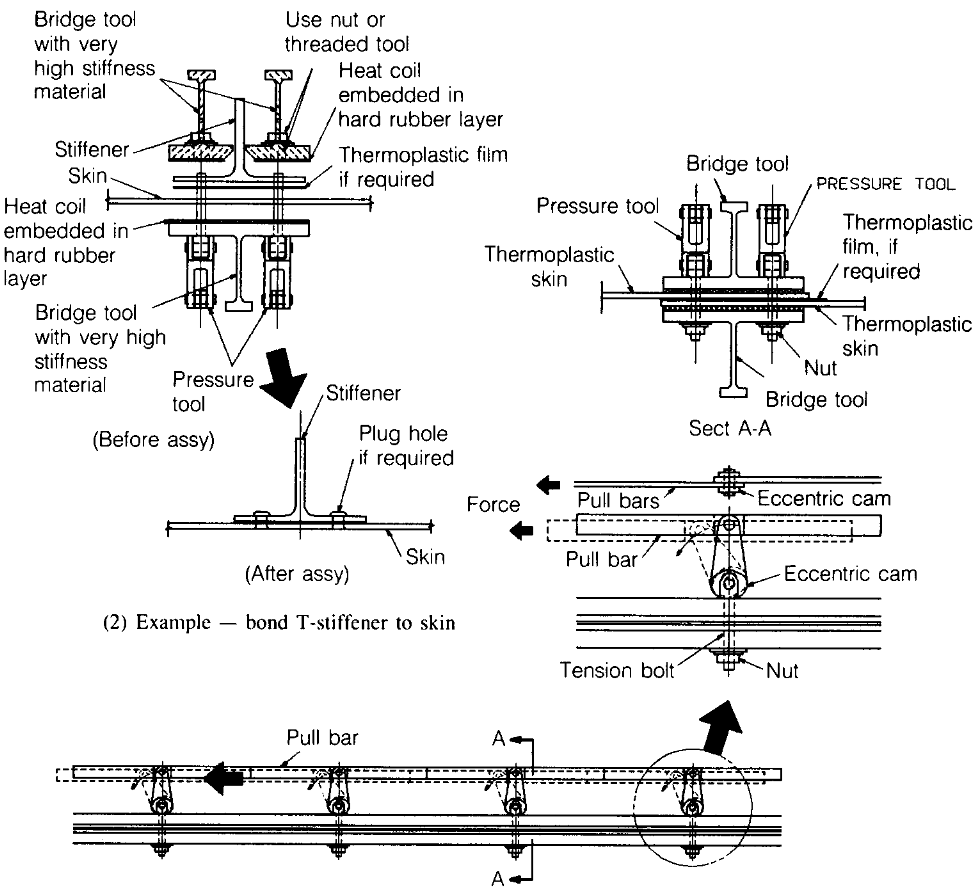
By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.4.3 Canopy Frame Fabricated By Robotic Tape Winding Placement Process Concept

A proof-of-concept of the full-scale forward segment of a thermoplastic composite canopy frame (see Fig. 12.4.2 and photo shown on the front page of this book) was fabricated by hand layup to demonstrate this concept. It achieved 50% weight reduction and manufacturing costs compatible with the metal counterpart. To attack the goal of lowering manufacturing costs, a proposed robotic/automated tape winding placement technology,shown in Fig. 12.4.3, should be considered to produce the future full-scale composite canopy frame.

12.5 ASSEMBLY CONCEPT

Thermoplastic composite structures can be reconsolidated by applying appropriate heat and pressure onto the part. The mechanical tool shown in Fig. 12.5.1 allows out-of-autoclave joining and is designed to be used for applying both pressure and heat locally along structural joint areas. A compatible thermoplastic film with a lower melting temperature may need to be inserted between the contact surfaces of the joint. Many joining methods such as local heating, induction welding, resistant welding, etc. could be considered to co-consolidate the contact surfaces to obtain a sound bonding joint.



(1) Typical assembly joining by local heating, induction welding, etc (bridge tools not shown)

By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.5.1 Out-Of-Autoclave Assembly Joining

12.6 OTHER CONCEPTS

(1) Countersunk dimpled holes:
A minimum panel thickness at mechanical fastener joint locations is required due to local countersunk fastener heads. A method of dimpling a hole on a solid skin or SynCore sandwich panel of thermoplastic composites can be developed. During the dimpling operation, local heat and pressure are required. Fig. 12.6.1 illustrates this concept.

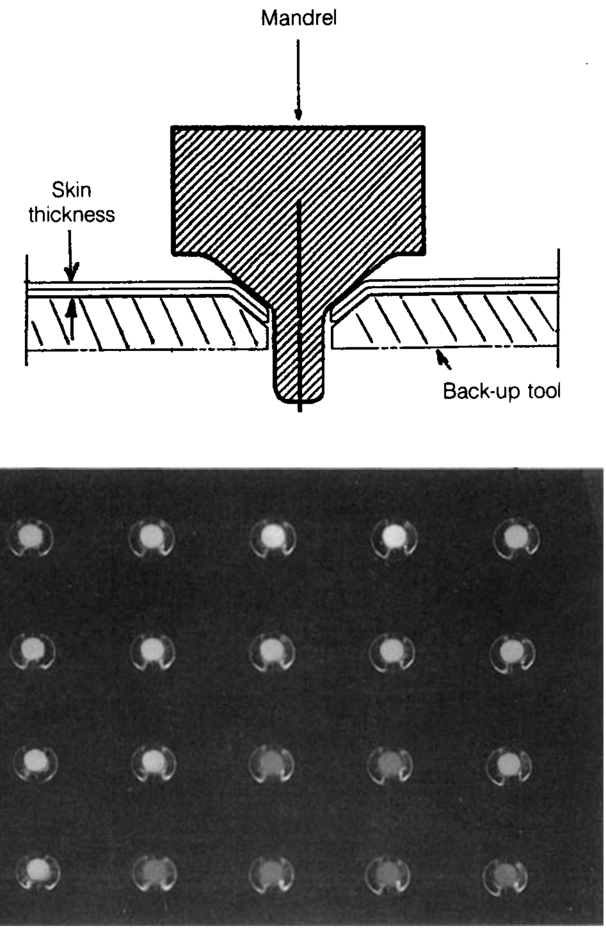
(2) Fits-in frame:
This concept utilizes stretchable thermoplastic material which is embedded into the frame at several selected locations. The frame is expanded by using clamp device and local heat source, as shown in Fig. 12.6.2.

(3) Step-hollow-grid:
This concept consists of a basic skin and a preformed step-over-grid-skin, which is superformed, as shown in Fig. 12.6.3. The depth and width of the grid-skin are either constant or variable as needed throughout the entire panel. Transverse and longitudinal reinforcements are cocured or bonded onto the skin-grid to provide additional strength and/or stiffness. This concept can be applied on door design as illustrated in Fig. 12.6.4.

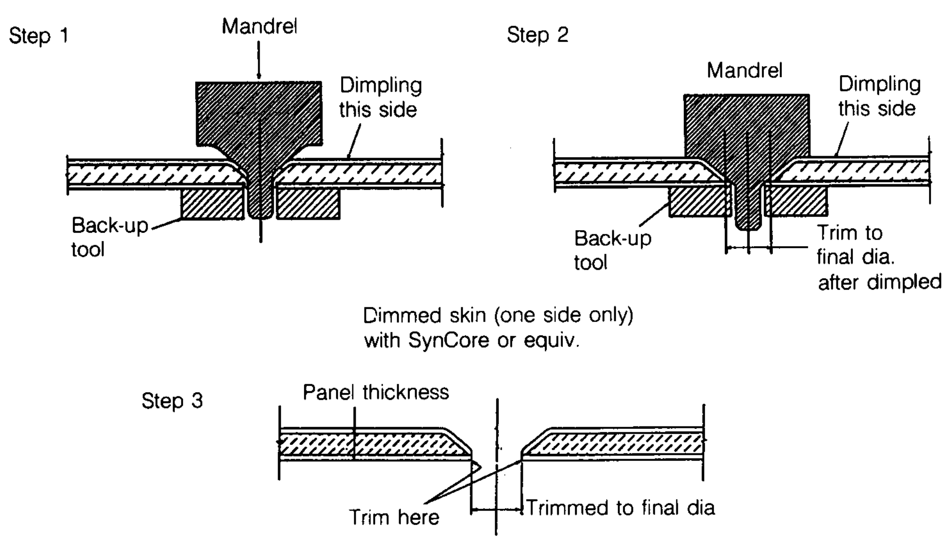
(4) Even-hollow-grid:
This concept consists of a basic skin and a cross-hat-grid skin which are bonded (thermosets) or dual-resin bonded (thermoplastics) together to form a wing or fuselage panel (see Fig. 12.6.5). All grid caps (longitudinal and transverse directions) are continuous through grid intersections.

(5) Fastenerless construction:
This concept is an ideal method for use on control surface structures since it does not used a single fastener (see Fig. 12.6.6). This construction consists of a series of modular boxes which are wrapped around a pre-formed washout mandrel. The modular boxes are packed together and the outer skin is wrapped around the entire assembly. The assembly is cocured in one operation. Cutouts on the spar web are required not only to allow washout of the interior mandrel materials but to allow for inspection and future maintenance.

(6) Smart structures:
This concept represents a range of applications with one characteristic — the structure is more than just a framework designed to passively support the functional parts of the system. In airframe applications, the smart structure concept may be used at various levels of smartness. For example:
   (a) Aircraft antenna systems have panels which incorporate built-in conformal electronics for transmitters, receivers and other sensors as "smart skins". The skin panel is designed to serve both as a structural member and as an electronic component (see Fig. 12.6.7).
   (b) Structural components can be fabricated with an array of built-in fiber optic sensors to monitor processing conditions during manufacturing (see Fig. 12.6.8). In the finished part, changes in passive signals from the same embedded optical system can warn of either service or battle damage to prevent fatally damaged aircraft from continuing to fly.



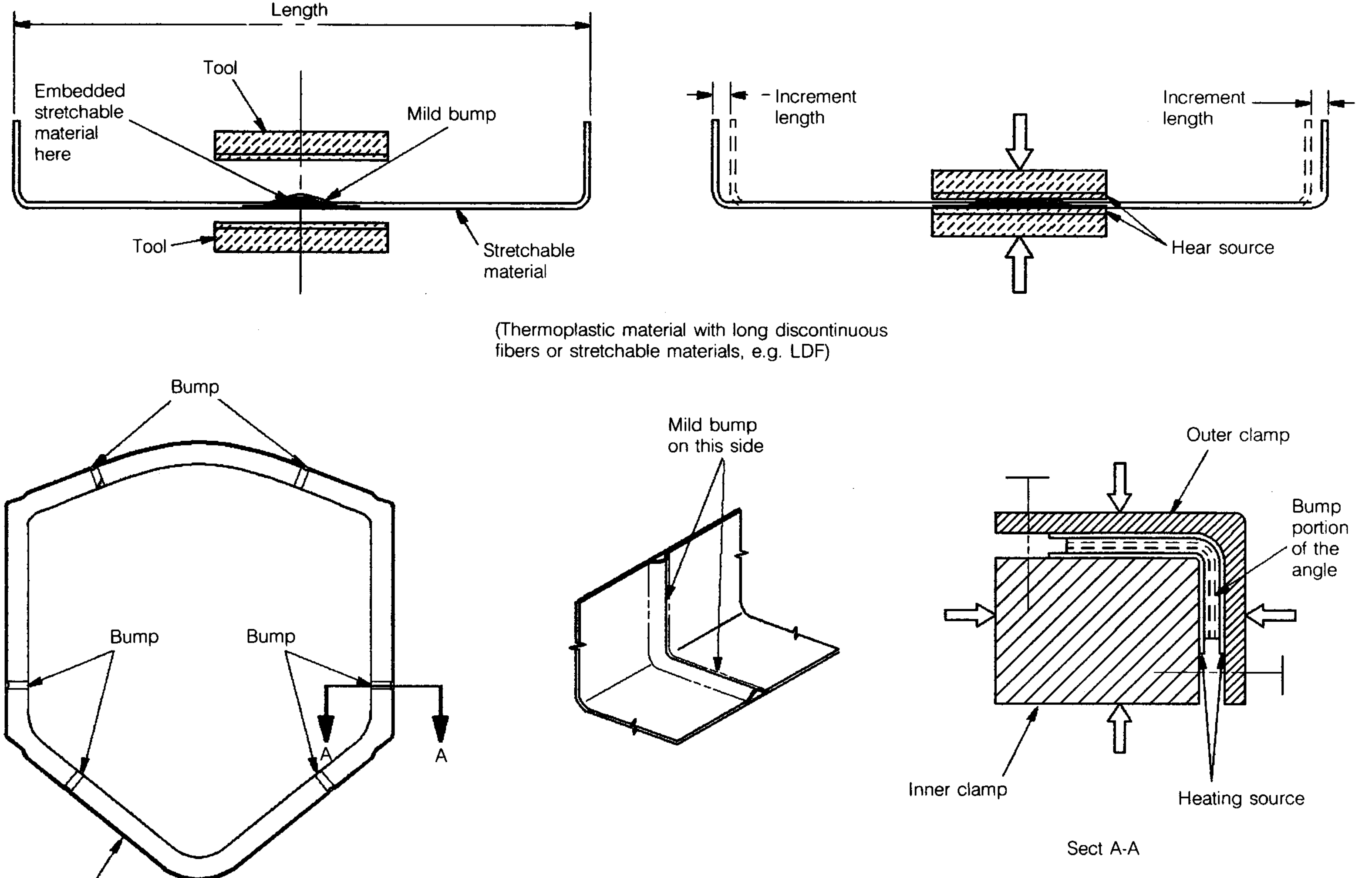
(a) Dimpled skin



(b) Dimpled sandwich panel (SynCore)

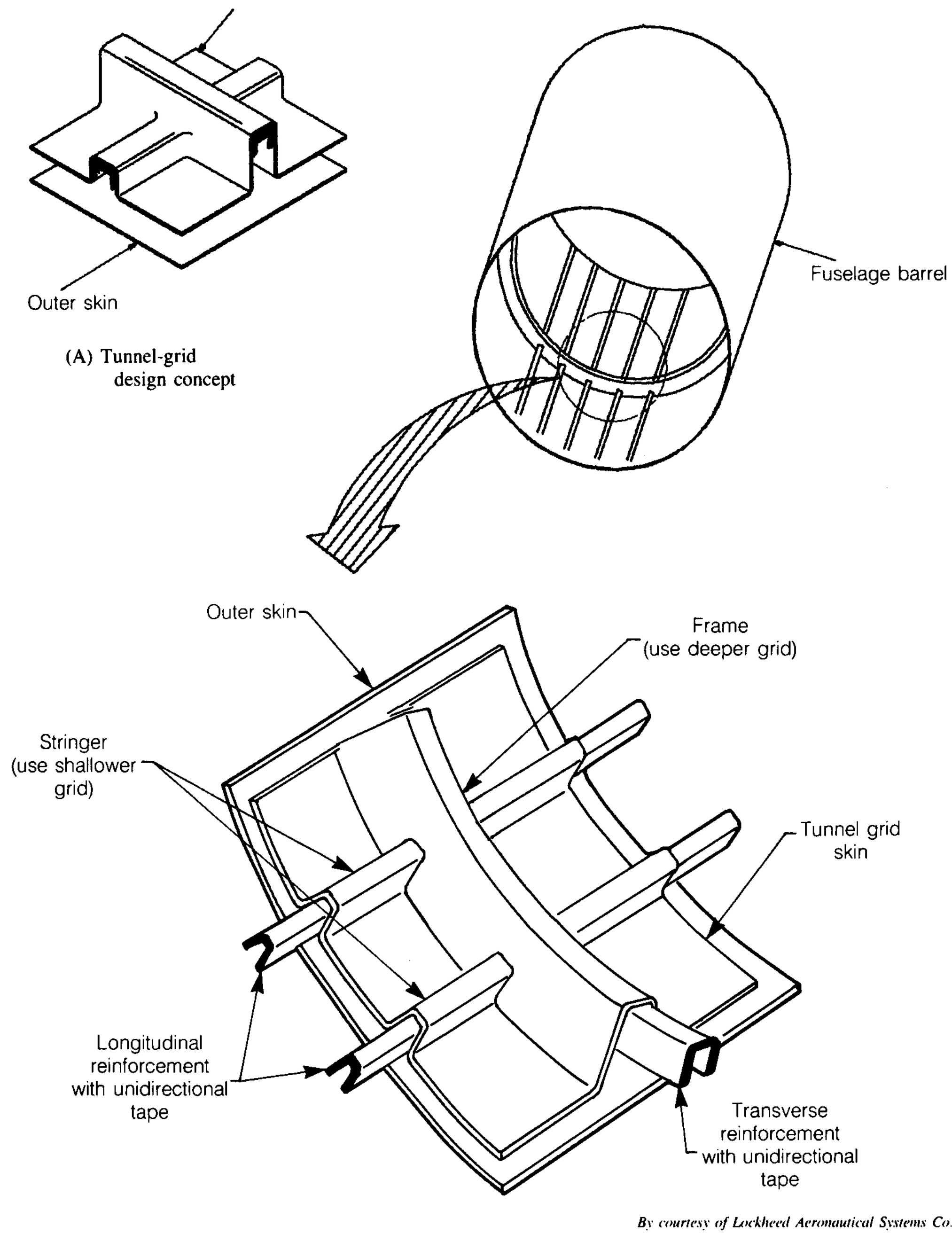
By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.1 Dimpled Panel (Thermoplastics)



By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.2 Fits-In Picture Frame



By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.3 Step-Over Grid Design Concert

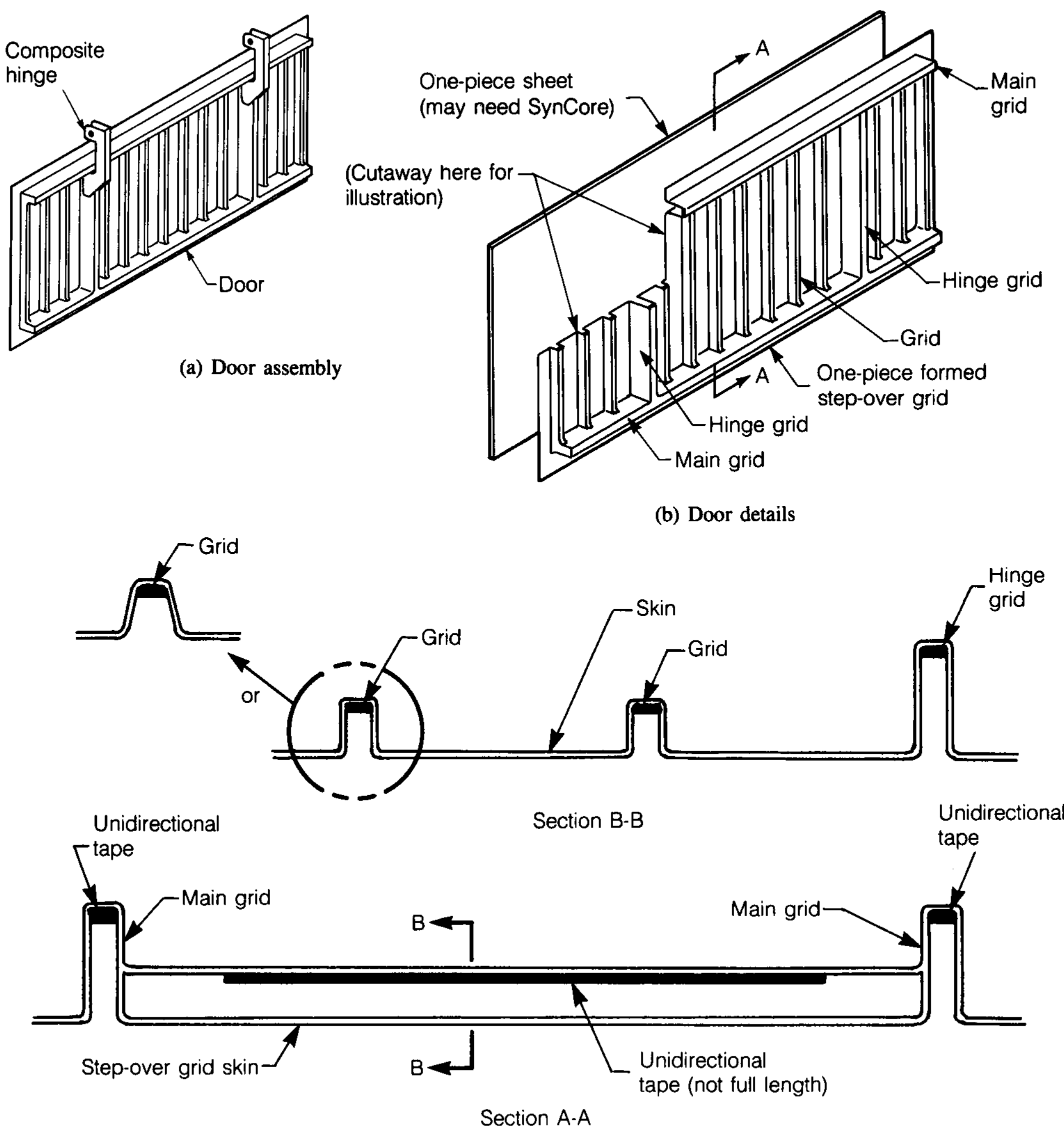
(7) Pre-stressed integrally-stiffened panel:

The theory of composite pre-stressed structures is the same as conventional pre-stressed concrete construction. In composite structures, very high strength tensile fiber tows (similar to steel bars used in concrete) such as 700 ksi (4791 Mpa) are used. These fiber tows are embedded into the panel, as shown in Fig. 12.6.9, and this protection to ultimately allows utilization of the full strength of the tow materials which further reduces structural weight.

(8) 3-D fiber-link composite structure

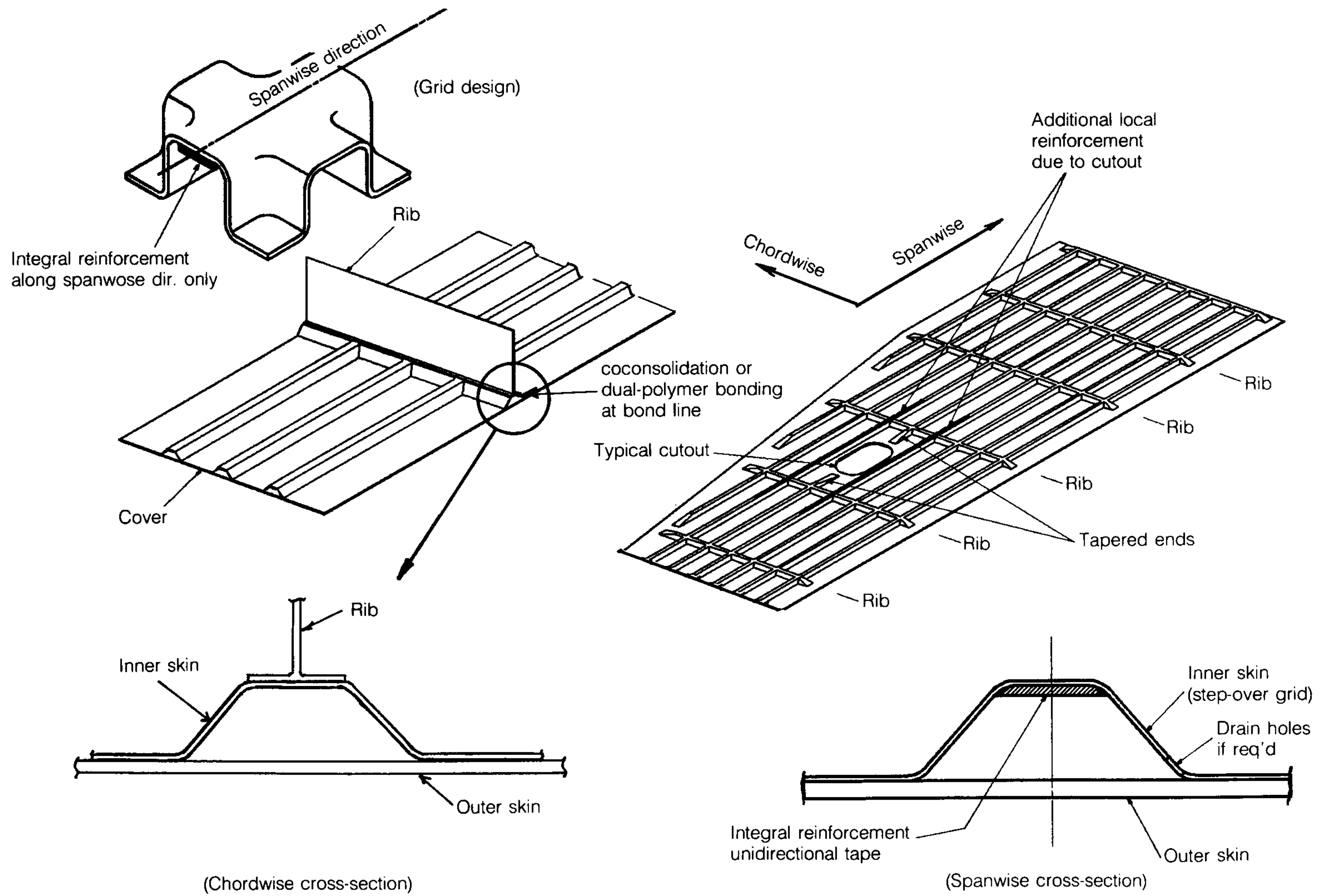
The marvelous handicraft weave method of rattan (see Fig. 2.5.25 in Chapter 2) is combined with resin transfer molding (RTM) to construct a sound composite structure which has all continuous fibers in three directions. This is a ''dream composite structure'' which provides the advantage of increasing not only load-carrying capability, but also impact resistance in composites and overcomes the weakness in today's composite technology.

Ultimately this method should rely on a robotic/automated technique to reduce the high costs which result from such a labor-intensive process. This concept should follows the same procedures as described for Very Complex Shapes (VCS) in Section 2.5 of Chapter 2).



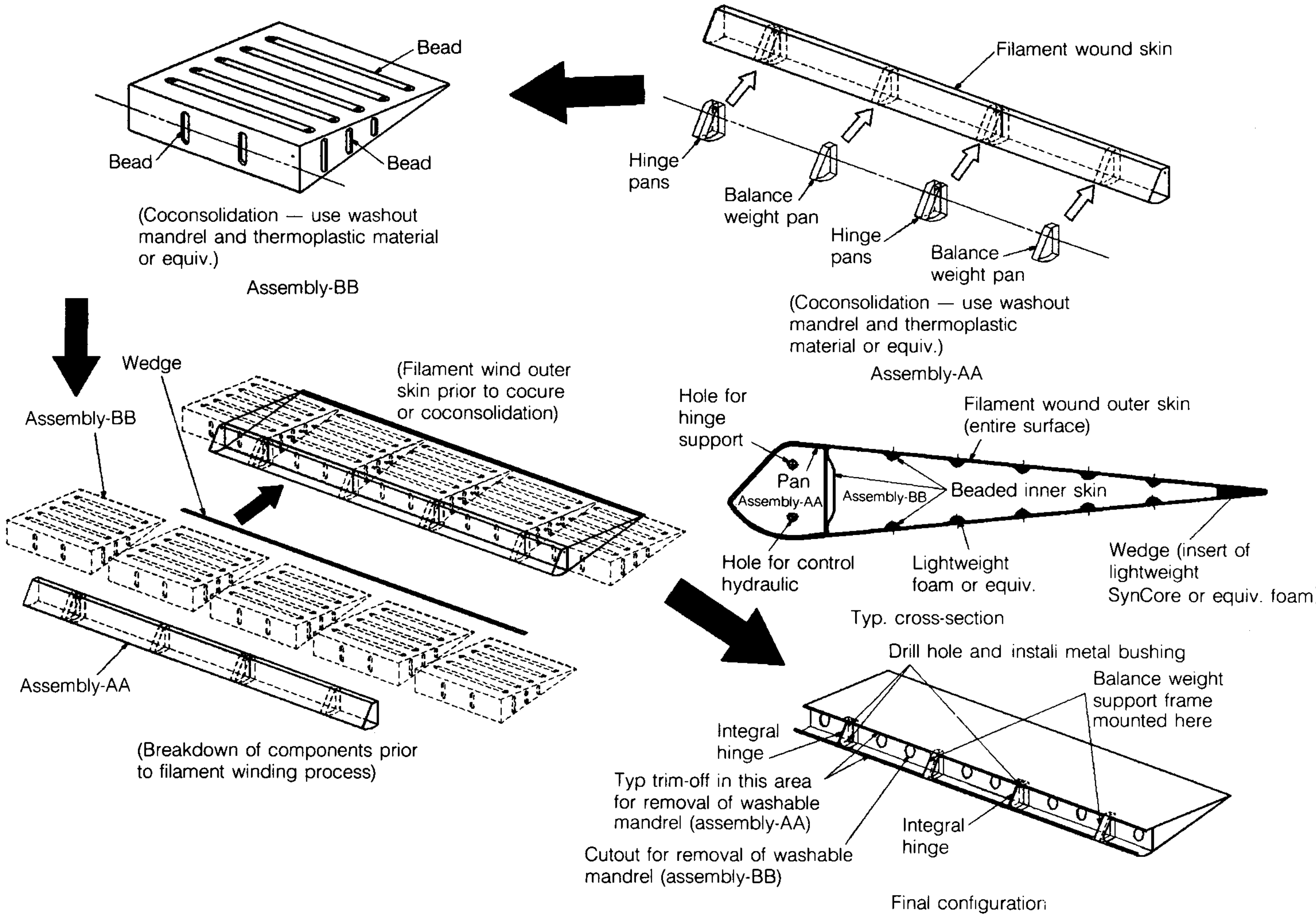
By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.4 Step-Hollow-Grid Door Design



By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.5 Even-Hollow-Grid Wing Cover



By courtesy of Lockheed Aeronautical Systems Co.

Fig. 12.6.6 Fastenerless Construction (Aileron Box)

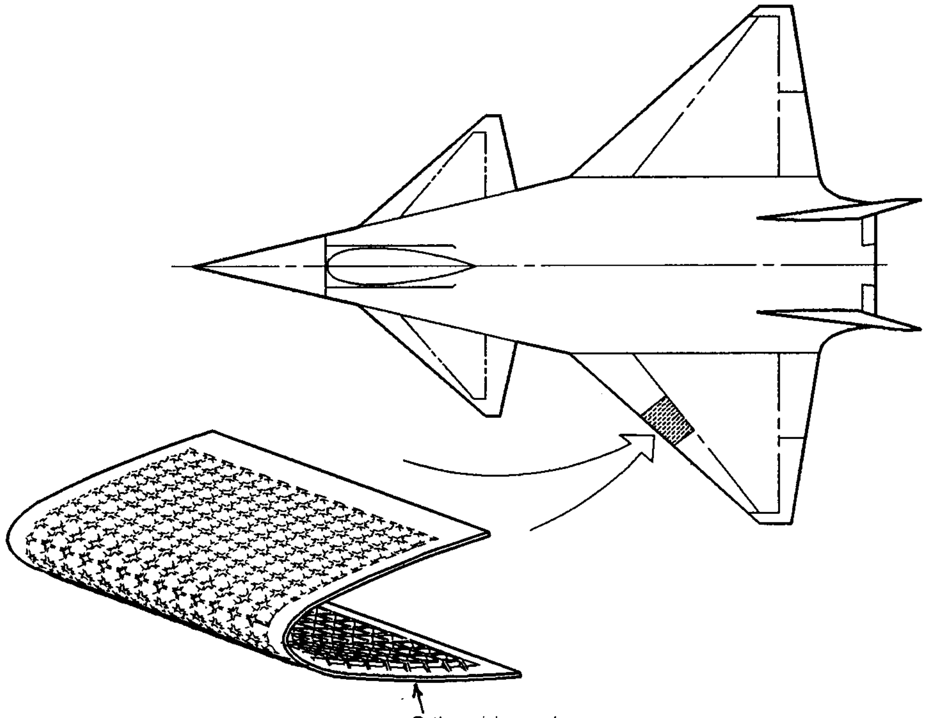


Fig. 12.6.7 Wing Leading Edge Orthogrid Panel And Avionics Integration (Smart skin)

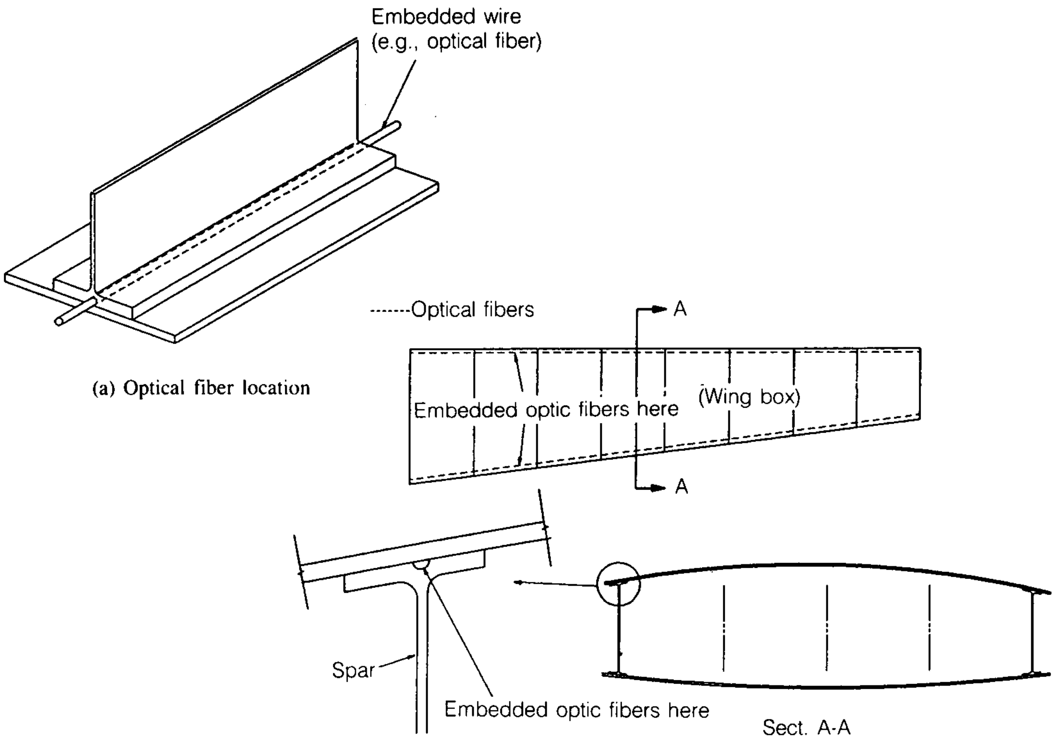
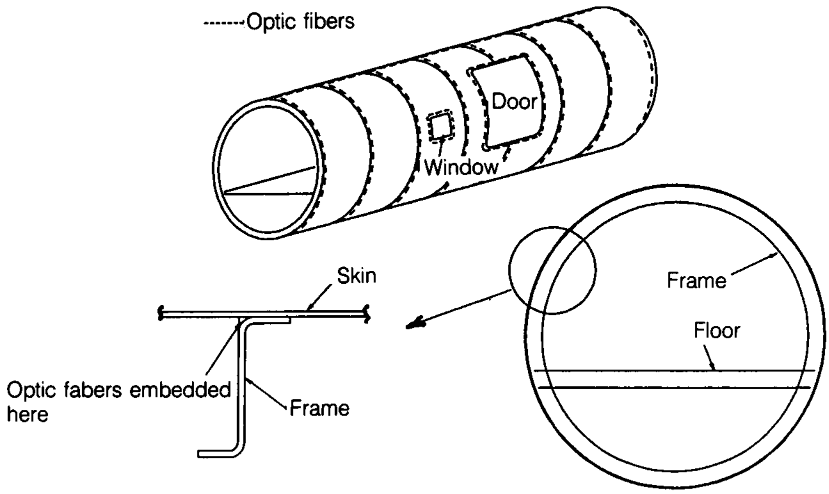


Fig. 12.6.8 Smart Structure Applications



(c) Fuselage

Fig. 12.6.8 Smart Structure Application (cont'd)

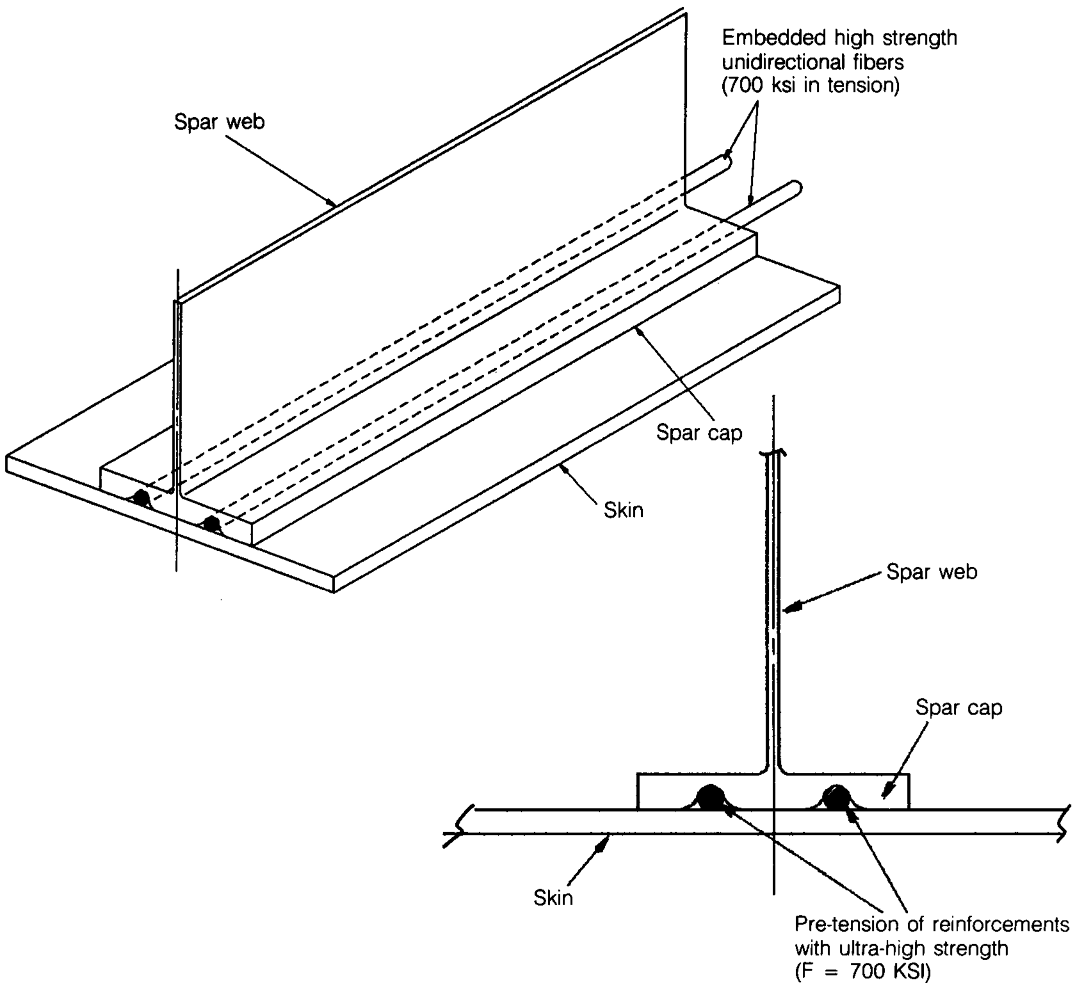


Fig. 12.6.9 Pre-Stressed Panel

12.7 GREEN AIRFRAME

Currently, many aeronautical engineers and researchers are working diligently toward the so-called 'Green Technology Airplane' to save structural weight to make the aircraft more fuel-efficient. The use of composite material is the obvious choice to achieve this goal, but structural weight saving is not the only means, because every airframe structural design has to meet many other requirements for certification. However, the use of composite material alone is not enough to reach this goal; the adoption of innovative design concepts could make the 'Green Airframe' dream come true. Chapter 12.0 presents innovative design approaches and Chapter 12.7 contains additional design concepts for commercial airplanes, giving readers some suggestions and recommendations as a starting point. There are still some other weight savings which are not directly related to airframe structures, but it is worthwhile for airlines to consider them for operating a 'Green Airplane' as well as for cost savings.

Airframe design is a team effort involving compromise with other disciplines (see Fig. 12.7.1); the compromise requirements vary with different types of airplane, e.g., commercial transport, military transport, commuter airplane, business jet-plane, fighter aircraft, general aviation, etc. With either metallic or composite airframes, the most important issue is how to certify the airframe; composite airframes, however, require for more effort to accomplish this task.

All the design approaches presented in this section are concepts only, and the final details necessary to build a prove-of-concept prototype to verify and accomplish feasibility may require further research.

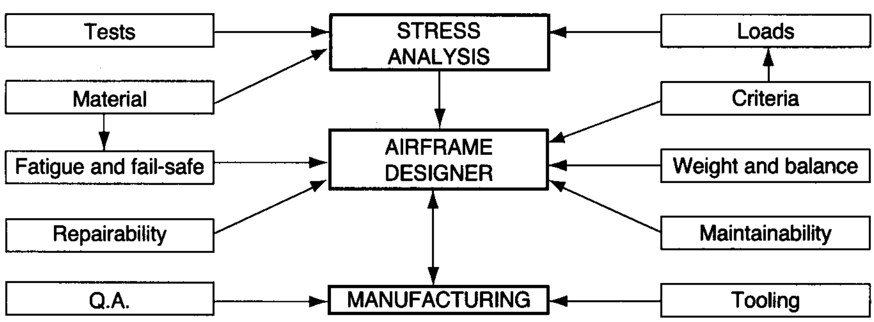


Fig. 12.7.1 Successful Team Work is the Foundation of the 'Green Airframe'

"Co-design"

Co-design is a term coined by the author to describe the concept of choosing the right material used in the right place to accomplish the right task.

(1) Material selection versus type of load:

(a) For compression load case:

- Critical failure mode: buckle
- Compression stress: F_{cy} (metal max. crippling stress)
- Impact damage tolerance (composite)
- Best material candidate is **aluminum**

(b) For tension load case:
- Critical failure mode: crack (metal)
- Tensile fatigue stress: $F_{fatigue}$ (metal)
- Best material candidate is **composite**

(c) For shear load case:
- Critical failure mode: shear-off or buckle
- Shear stress: F_{su} (metal shear-off and/or buckle)
- Best material candidate is **composite** with increased use of ±45° plies

(d) For crash landing case:
- High crash energy absorbability
- Best material candidate is **aluminum**, **titanium**, **steel**, etc.

(2) The co-design case studies:

(a) Transport fuselage:
- Use mostly composite material for upper-half shell and frames, stringers, etc. to improve fatigue
- Use mostly aluminum material for lower-half shell and frames, stringers, etc. for compression loads and crash landing
- A special design is required at the longitudinal joint at the location between upper-half and lower-half shells to resolve thermal expansion problem (see Fig. 12.7.2)
- A thin titanium plate is needed between composite skin and aluminum skin to reduce thermal expansion problem and also to prevent galvanic corrosion

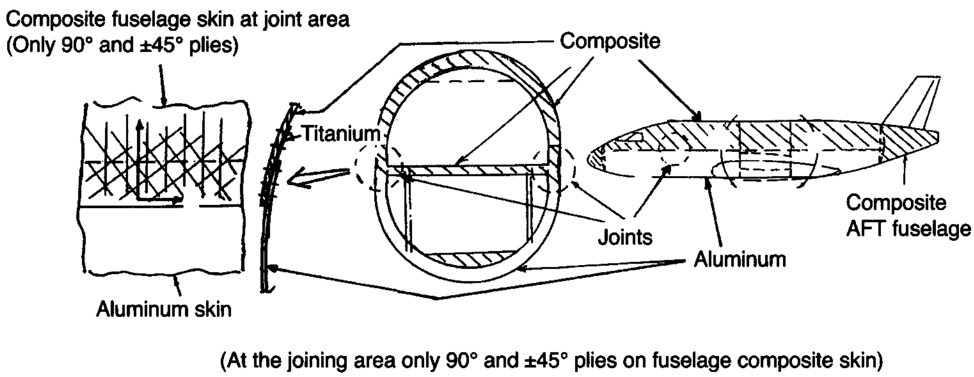
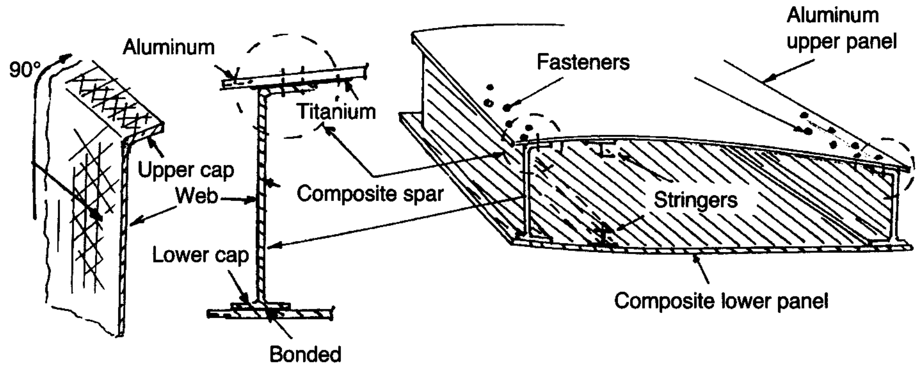


Fig. 12.7.2 Thermal Expansion Joint Along Sides of Fuselage Body

(b) Wing box:
- Use aluminum material for upper panel – Primarily compression load and lightning strike critical
- Use composite material for lower panel – Primarily tension load fatigue critical
- Shear load is critical for spar using composite material by laying mostly of ±45° plies and small amount of 0° plies (or no 0° plies at all) at attachment location to upper aluminum panel to compensate for the span-wise thermal expansion compatibility between them (see Fig. 12.7.3)

- A thin titanium plate is needed between upper cap of composite spar and aluminum upper panel to reduce thermal expansion and also to prevent galvanic corrosion



Upper spar cap area: only 90° and ±45° plies
Lower spar cap area: more 0° combined with ±45° and 90° plies
Spar web area: more ±45° combined with 90° plies and few 0° plies

Fig. 12.7.3 Thermal Expansion Spar Design

Honeycomb Panel Design

The honeycomb panel is the ideal and highest structural efficiency for airframe structures; however, it has many technical problems that need to be resolved. This section mentions several concepts, but further research and development is necessary to reach the airframe engineer's ideal.

Improving honeycomb strength (see Fig. 12.7.4):

- Increase shear and tension strength between surface skin and core
- Dip core into special resin and dry all drips by laying on a heated plate tool
- These drips (or fillets) will improve strength between skin and core
- May consider filling very light-weight density foam into core and use as insulation to lower thermal conductivity for the fuselage barrel panel

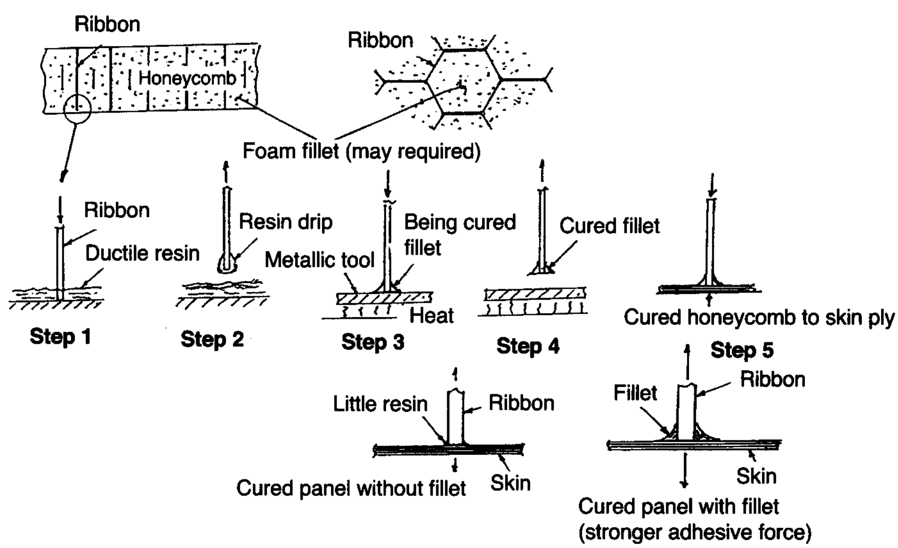


Fig. 12.7.4 Process for Improving Strength of Honeycomb Panel

(1) Wing box:

The wing box design concepts are also applicable to both horizontal and vertical tail box.

(a) Wing box upper panel (see Fig. 12.7.5):

- Pre-cured rectangular vent tubes, imbedded in the honeycomb core panel and co-cured together
- All vent tubes are designed as structural members to pick-up load
- No additional tubes (e.g., hat-sectional stringers in upper wing skin-stringer panel) required for fuel ventage

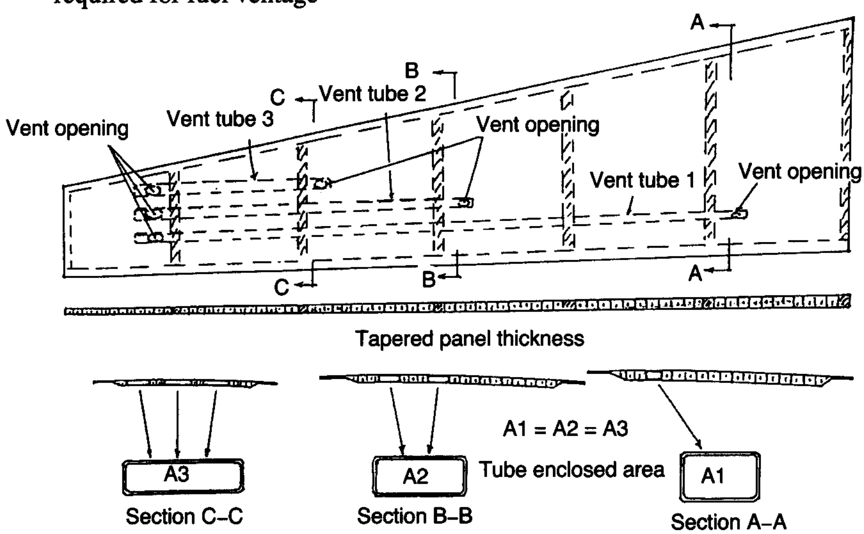


Fig. 12.7.5 Wing Upper Honeycomb Panel with Imbedded Vent Tubes

(b) Wing box lower panel (see Fig. 12.7.6):

- Clamped-in pre-cured cover fabricated from honeycomb core panel
- Cover is designed to prevent fuel leakage and also for lightning strike protection

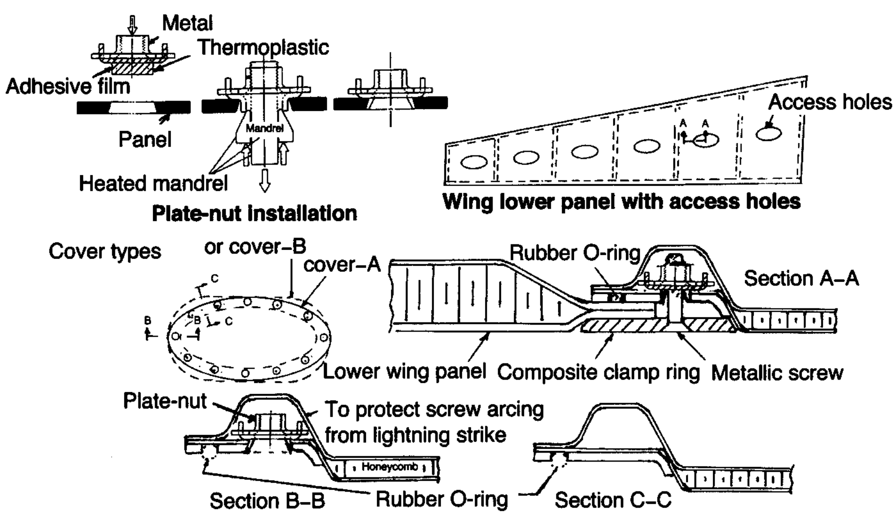


Fig. 12.7.6 Wing Lower Honeycomb Panel with Access Holes

(c) Attachments between wing panel and rib (see Fig. 12.7.7):

- Drill countersunk-like hole from outer skin of the panel and fill up with plotting compound in the honeycomb core, then install and bond a preform (or pre-cured) composite funnel-cap to the drilled hole by using adhesive
- Easily create funnel-cap by forming a thermoplastic sheet with chopped fibers in it
- Fill up with ductile potting compound (mixed with a small amount of chopped fibers) into the funnel-cap from the panel outer skin
- Potting compound is held in place by a small composite retainer to prevent it from falling away. The potting compound will automatically prevent the lightning strike problem.
- Easily replace fastener by drilling out potting compound and unscrew the fastener (e.g., Hi-Lok or Eddie Bolt), then re-fill the new potting compound and new retainer after replacing the fastener

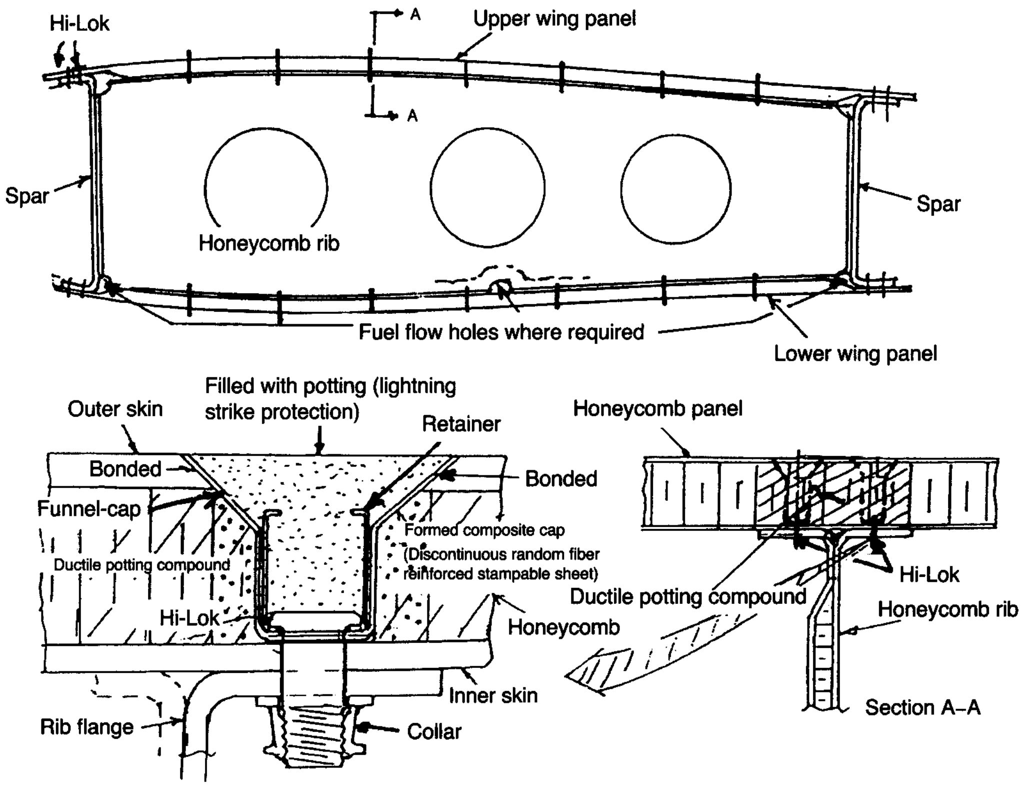


Fig. 12.7.7 Attachments between Wing Panels and Rib

(d) Composite wing root joint (see Fig. 12.7.8):

- Pre-cured double-cruciform fitting (woven) using composite material as shown in Fig. 12.7.8, Fig. 12.7.9 and Fig. 12.7.10

- Honeycomb panels (inboard and outboard panels) are secondary bonded to the double-cruciform fitting
- May use multiple rows of carbon straight tows (in both vertical and horizontal webs of the fitting) with taper as shown in Fig. 12.7.8

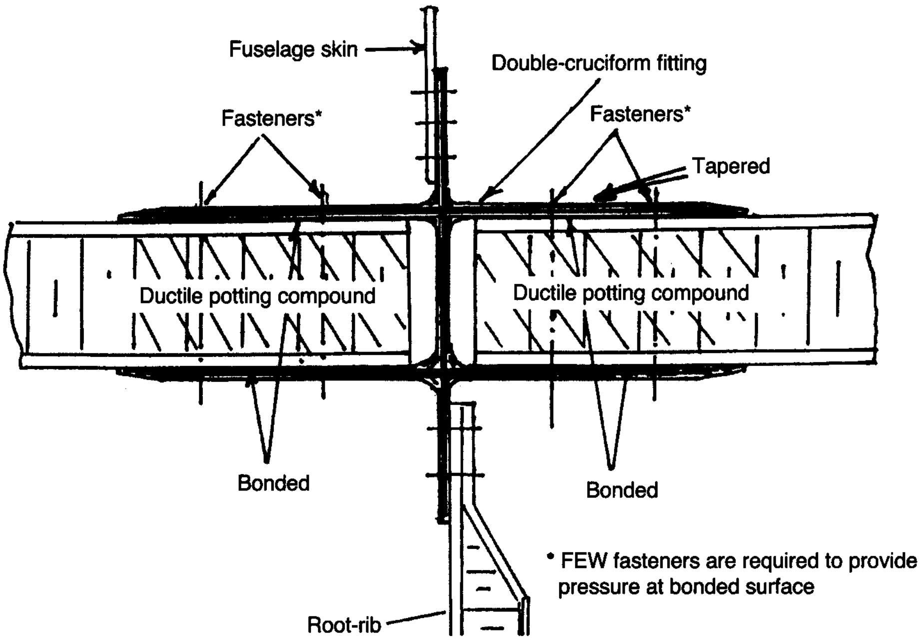


Fig. 12.7.8 Composite Wing Root Joint

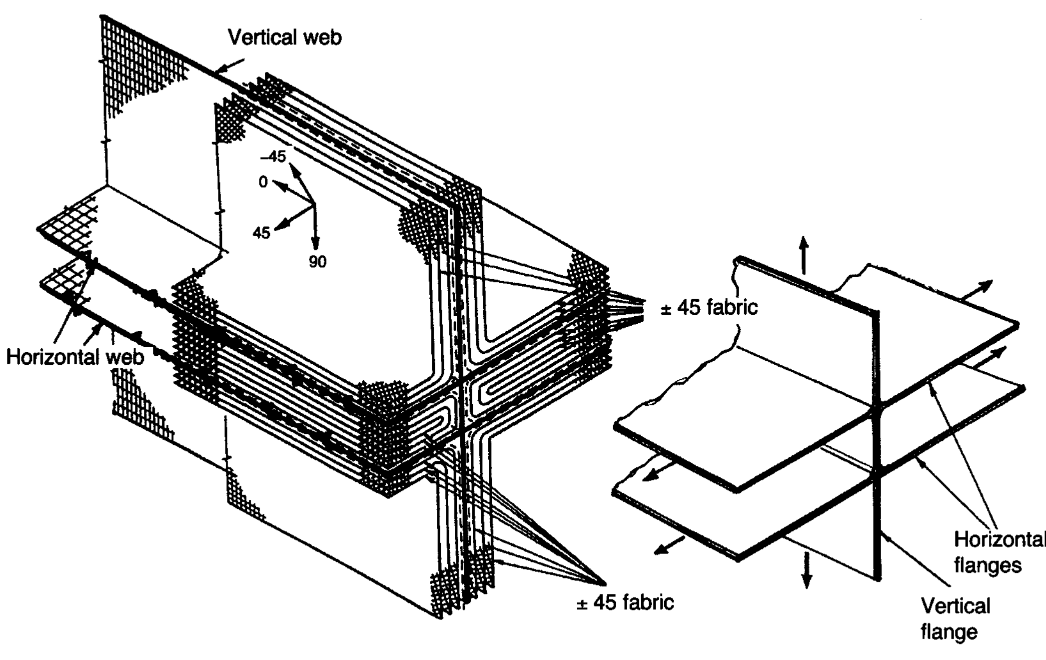


Fig. 12.7.9 Composite Double-Cruciform Fitting

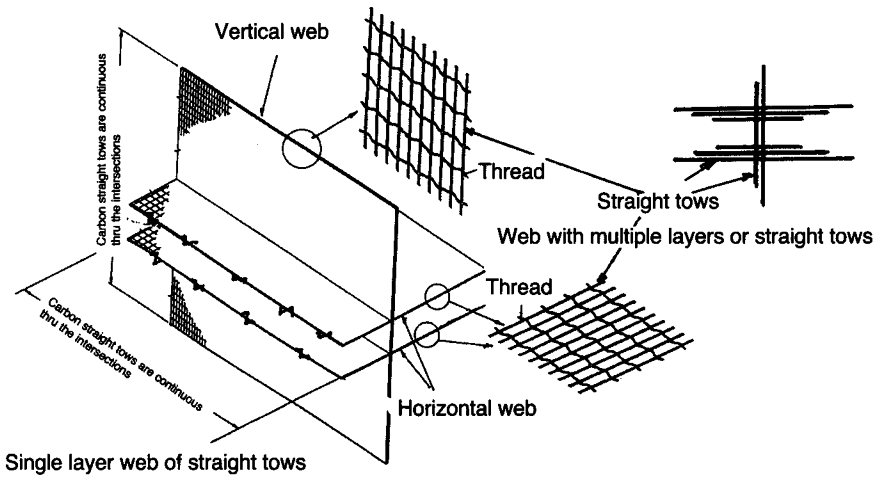


Fig. 12.7.10 Detail of Composite Woven Double-Cruciform

(2) Rectangular cutout corner design:

- Patch a pre-form with fiber tows (prepreg fiber-wrap) at every corner to reduce stress concentration and co-cured together with panel (e.g., wing box access holes, fuselage window, door, etc.)
- Use typical reinforcement around a honeycomb barrel fuselage window with fiber-wrap and RTM J-frame for mounting window pane, as shown in Fig. 12.7.11

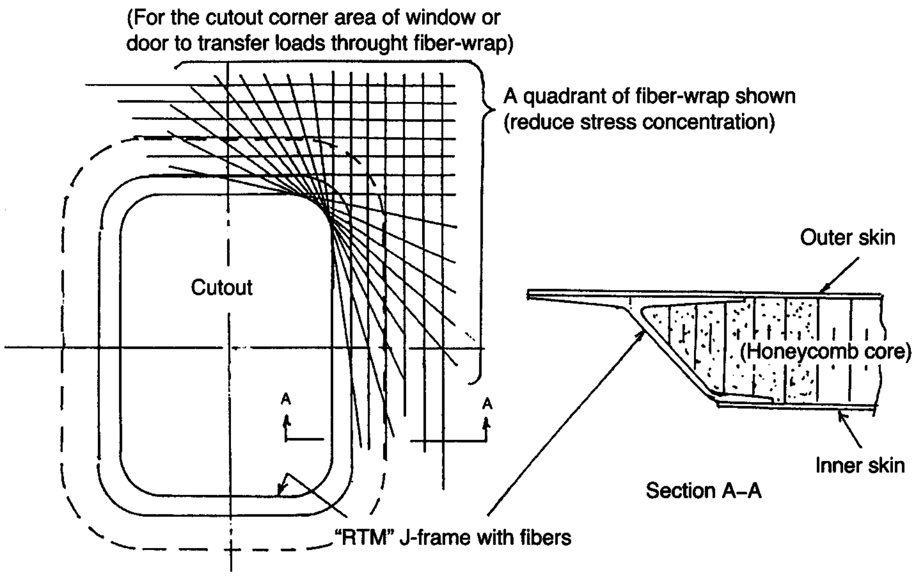


Fig. 12.7.11 Typical Fiber-wrap at the Cutout Corners of a Fuselage Window J-frame

(3) Fuselage barrel panel:

The honeycomb panel application provides the most structural efficiency for the fuselage barrel.

- This concept has been illustrated previously in Fig. 12.3.2, except that the fuselage barrel is replaced with honeycomb panels
- Honeycomb barrel construction is shown in Fig. 12.7.12
- Fill with light weight ductile foam (or potting compound) into the core, which will lower the thermal conductivity for the honeycomb fuselage barrel panel. This foam replaces the traditional fuselage insulation blankets.
- No fasteners between frame and barrel panel; they are integrally co-cured together to avoid creating tension load between frame and barrel panel due to the fuselage cabin pressurization
- Fuselage longitudinal axial loads are taken up by straps (mainly 0° ply tapes), which are pre-stressed prior to bonding on honeycomb fuselage barrels in the final assembly (see Fig. 12.7.13)

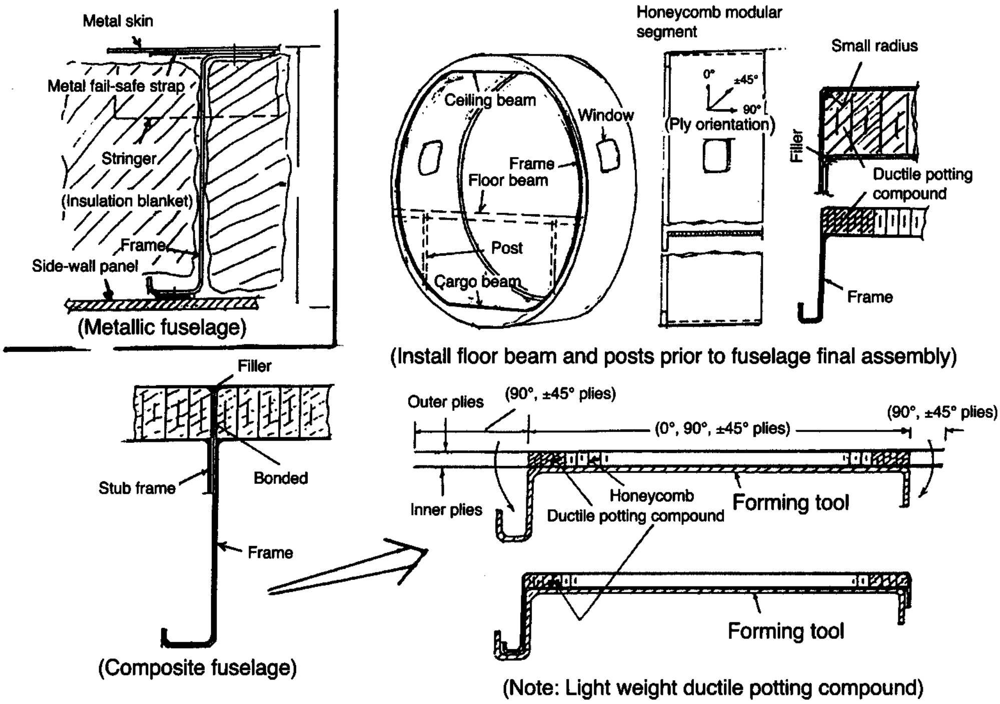


Fig. 12.7.12 Construction of Fuselage Honeycomb Barrel Panel

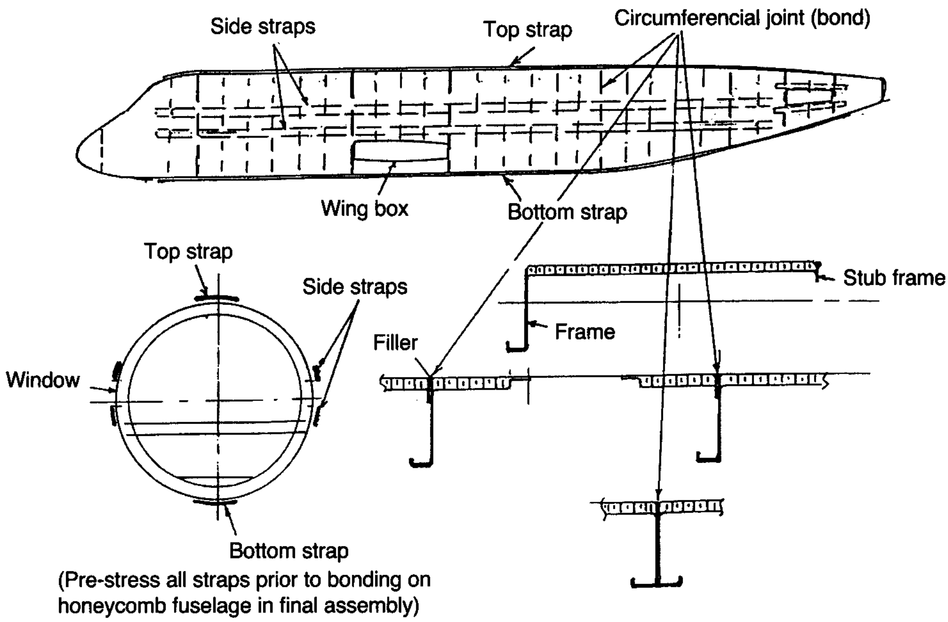


Fig. 12.7.13 Fuselage Final Assembly

Composite Bow-tie Connector

Conventional tension fitting is machined from metal, as shown in Fig. 12.7.14(a), which is currently being used on many composite structures. Each fitting is mechanically fastened onto the basic structure by many fasteners, producing stress concentration. Tension load transfer is through a tension bolt between these two tension fittings which can be detachable by unscrewing the tension bolt (e.g., as required by a fighter wing box). But many other tension fittings are designed without this requirement. In this case tension fittings can be replaced by a composite bow-tie connector, as shown in Fig. 12.7.14(b).

- Generally, bow-tie connectors are not removable because they are bonded on the basic structures instead of fasteners
- No need for any fasteners, including the tension bolt
- This is the most efficient joint because there is no stress concentration area from fasteners, in addition to being light weight

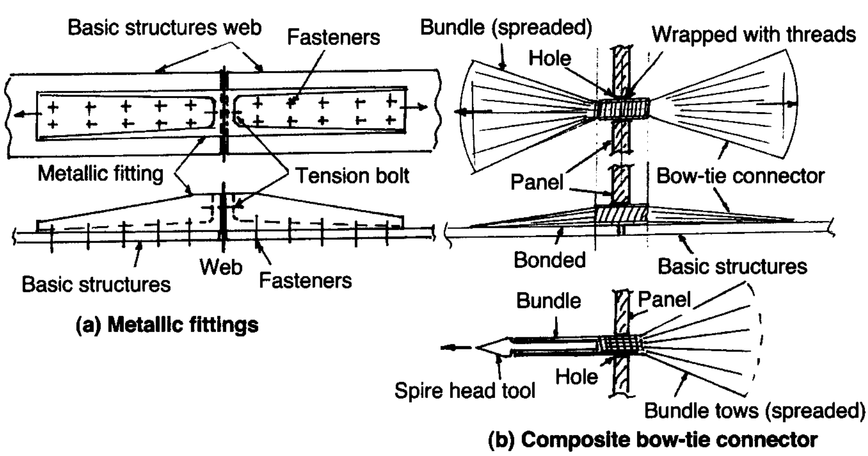


Fig. 12.7.14 Composite Bow-tie Connector

Other Relevant Weight Saving

Currently, the Green Technology Airplane emphasizes both the fuel efficient engine and airframe structure weight savings, resulting in less emissions pollution from engines. The airplane manufacturer may be bearing too much of the burden to achieve this result. Both the airlines and passengers should follow the 'Share Responsibility' ideal, which is the other part of significant aircraft weight saving.

The 'Share Responsibility' (suggestions and recommendations):

(1) Fuselage warehouse interior:

- Warehouse stores are popular shopping places; most people find this type of shopping environment acceptable
- Warehouse interior will not have fancy or luxuriously decorated side-wall and ceiling panels, saving both weight and cost, just the bare bones of the airframe structures, similar to the interior of a military transport
- Warehouse interior is feasible for short haul transport with flight range around 3 hours
- Warehouse interior sacrifices passenger comfort but does not compromise airplane safety. Most passengers may choose low airfare over comfort
- Select light and thin (less pages) airline propaganda magazine, or completely eliminate them, to save weight
- Use regular fixed chair (non-adjustable seat back) to save weight and cost
- Take away overhead reading lights, and small air conditioning outlets, and all entertainment systems, saving tremendous numbers of wiring and tubes
- Take away heavy galley and food carts; instead, each passenger will receive a small bag containing a sandwich, with a bottle of water to replace onboard potable water tank
- Empty galley space could be converted to additional one or two more rows of passenger seats to increase airlines revenue

(2) Fuselage exterior:

Use less paint on fuselage exterior to save weight; use paint only for airline logo and identification numbers

(3) Aircraft fuel management:

- Consider using less fuel for each flight segment instead of full tank of fuel to cover many flight segments
- Fuel management can be easily controlled by modifying onboard computer system

(4) Onboard life-raft(s)

- Take off life-raft(s) if the aircraft is not flying over water
- Some airlines may already be practicing this

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12.3 Chu, R. L. and Niu, M.C.Y., "Innovative Design and Fabrication of Composite Canopy Frames", Ninth DOD/NASA/FAA Conference on Fibrous Composites in Structural Design, November 4-7, 1991.

12.4 Leonard, L. ''Smart Composite: Embedded optical Fibers Monitor Structural Integrity'', ADVANCED COMPOSITES, Mar/April, 1989. pp. 47-50.

APPENDIX A

Commonly Used Conversion Factors (English units Vs. SI units)

LENGTH:
- mil (0.001 inch) = 25.4 micron
- in = 2.54 cm
- ft = 0.3048 m
- ft = 0.333 yd

AREA:
- in^2 = 6.452 cm^2
- ft^2 = 0.0929 m^2

VOLUME:
- in^3 = 16.387 cm^3
- ft^3 = 0.02832 m^3

FORCE AND PRESSURE:
- psi = 6.8948 kpa
- ksi = 6.8948 Mpa
- Msi = 6.8948 Gpa
- Hg(32°F) = 3.3864 kpa

MASS:
- lb = 0.4536 kg

DENSITY:
- lb/in^3 = 27.68 g/cc
- lb/ft^3 = 0.6243 kg/m^3

TEMPERATURE:
- °F = (9/5) °C + 32

THERMAL EXPANSION:
- in/in °F $\times 10^{-6}$ = 1.8 K$\times 10^{-6}$

THERMAL CONDUCTIVITY:
- Btu × in/hr × ft^2 × °F = 0.1442 (W/m)K

IMPACT ENERGY:
- ft-lb = 1.3558 J

COMMONLY USED SI PREFIXES:
- G — 10^9
- M — 10^6
- k — 10^3
- μ — 10^{-6}

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X

Y

Z