

Module10: Special Topics

Learning Unit-1:

M10.1 Mechanical Testing of Composites

The **mechanical testing** of composite structures to obtain parameters such as **strength** and **stiffness** is a time consuming and often difficult process. It is, however, an essential process, and can be somewhat simplified by the testing of simple structures, such as **flat coupons**. The data obtained from these tests can then be directly related with varying degrees of simplicity and accuracy to any structural shape. The test methods outlined in this section merely represent a small selection available to the composites scientist. Some, such as the **tensile coupon test**, are widely recognised as standards, whereas there are dozens of different tests for the measurement of shear properties.

M10.1.1 Tensile Testing

Tensile testing utilizes the **classical coupon test geometry** as shown below and consists of two regions: a central region called the **gauge length**, within which failure is expected to occur, and the two end regions which are clamped into a **grip mechanism** connected to a test machine.

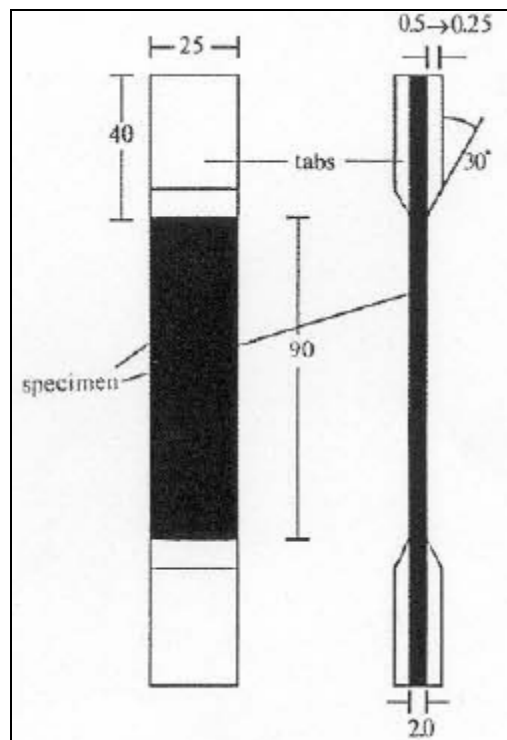


Figure M10.1.1 Typical tensile composite test specimen (all dimensions in mm)

These ends are usually tabbed with a material such as aluminum, to protect the specimen from being crushed by the grips. This test specimen can be used for **longitudinal, transverse, cross-**

ply and angle- ply testing. It is a good idea to polish the specimen sides to remove surface flaws, especially for transverse tests.

M10.1.2 Compressive Testing

This is much more problematical. The results obtained are essentially dependent on the type of compression fixture used. Also, the gauge length is conical, as if it is too long, the specimen will **buckle and flex**, resulting in premature failure. If it is too short, then the proximity of the tabs will adversely affect the stress state, resulting in artificially high values. The most widely used compressive test technique is the **Celanese fixture**, shown below. Cylindrical in design, a small specimen sits within a set of trapezoidal grips, encased in collars and an alignment shell. The gauge length depends on the type of test material and varies between 12.7mm for longitudinal specimens and 6mm for transverse specimens. Again, it is a good idea to tab the specimens.

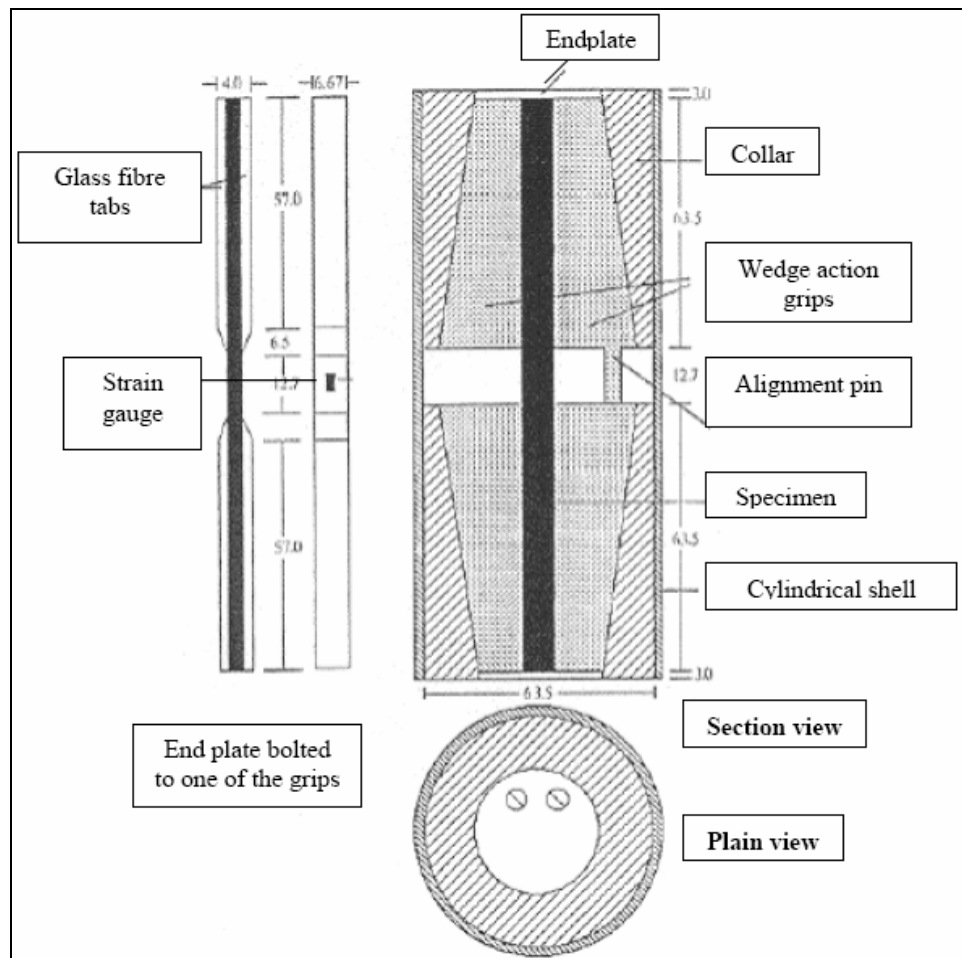


Figure M10.1.2 Celanese compressive fixture and specimen (all dimensions in mm)

M10.1.3 Intra-laminar Shear Testing

Most of the numerous **shear test** methods in existence measure intra-laminar shear properties, rather than inter-laminar ones. In theory, however, they should be the same in a perfectly consolidated material. A very popular test is the asymmetric **four-point bend test**; the essential features of which are shown overleaf. The specimen is 80mm long; 20mm wide and around 3mm thick, depending on the number of plies used in the test laminate. Two notches, 4mm deep, are cut where shown and application of the load will cause shear failure along the notch roots ('axial splits') followed by wholesale intralaminar shearing in the centre section. Strain gauges are bonded at $\pm 45^\circ$ in the specimen centre, enabling accurate **shear modulus** measurement (usually $G_{13} = G_{12}$ even in poorly consolidated materials)

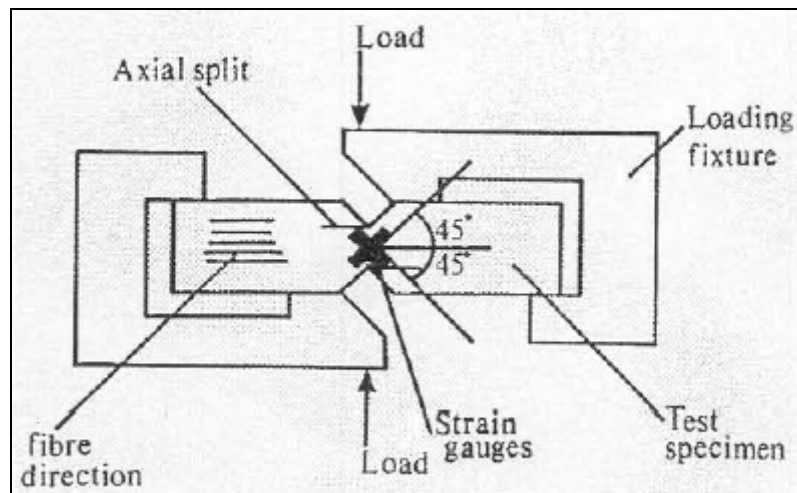


Figure M10.1.3 Schematic representation of the asymmetric four-point bend shear fixture

M10.1.4 Inter-laminar Shear Testing

The most common test for measuring **shear delamination** is the short beam shear test shown below, where a small specimen (<30mm long) is loaded in **three-point bending** until a delamination forms in the centre plane at one end of the specimen. 'P'.

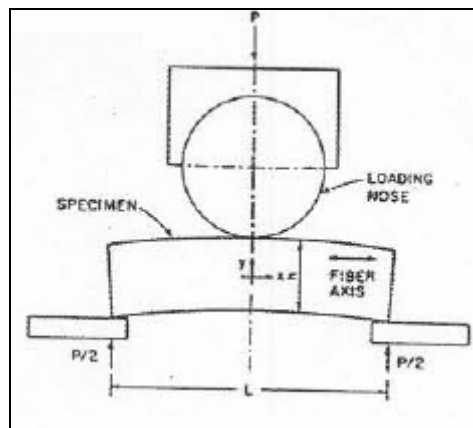


Figure M10.1.4 The short beam shear test

Such a test is difficult to optimize as failure will often occur by crushing under the central bending nose. In this test, the shear strength, τ_{12} is given as,

$$\tau_{12} = \frac{3P}{4bh} \quad (M10.1.1)$$

where 'P' is the applied load, 'b' is the specimen width and 'h' is the specimen thickness.

M10.1.5 (a) Mode-I Fracture Toughness

As a rule, Mode-I **delamination** in composites is measured using the **Double Cantilever Beam (DCB)** test method shown below, rather than the **Compact Tension test** geometry used for mode-I cracking of most other materials.

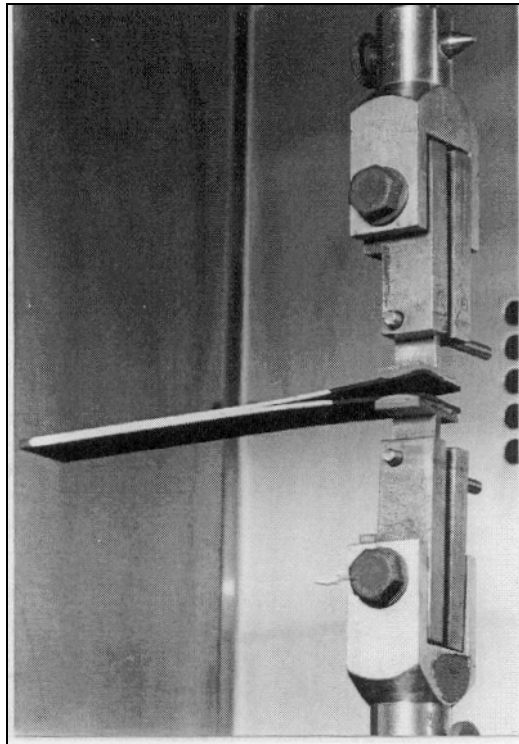


Figure M10.1.5 An in situ Double Cantilever Beam (DCB) test

The specimens are long, thin coupons (typically 150mm x 25mm x 3mm), tabbed at one end with aluminium hinges or 'T' tabs and with Teflon insert at the same end in the centre plane of the specimen to represent a delamination of known length. The test is a lengthy affair, as the specimen is loaded and subsequently unloaded many times after a small increment of **crack growth**. Referring to the load-displacement plot overleaf, crack growth is detected as a decrease in the trace. Nine load- unload curves are shown, with both the peak load (P) and the corresponding crack length ($a_1.....a_9$) recorded. From this, basic beam theory can then be used

to calculate a value of G_{Ic} -the mode I fracture toughness -as a function of crack length, according to equation (M10.1.2), as,

$$G_{Ic} = \frac{3P^2C}{2aw} \quad (M10.1.2)$$

where 'C' is the specimen compliance, given as the inverse gradient of the loading portion of each curve, and 'w' is the specimen width. This is the basic analytical equation of which numerous modifications exist.

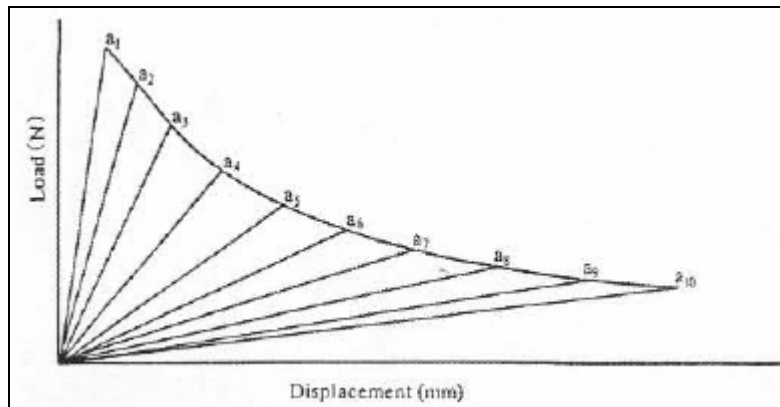


Figure M10.1.6 Typical load-displacement trace for the Double Cantilever Beam (DCB) test

It is found in most composite materials that G_{Ic} will vary with crack length quite significantly. This phenomenon is known as the '**R curve effect**', and is a critical design parameter.

M10.1.5 (b) Mode-II Fracture Toughness

Most mode II testing is conducted using the **End Notch Flexure (ENF)** test. This test uses identical coupons to the **Double Cantilever Beam (DCB)** test (minus tabs) and produces shear delamination under **three-point bending**, as shown Figure M10.1.7,

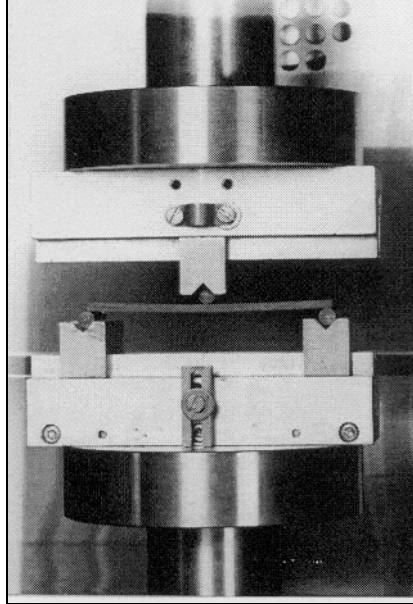


Figure M10.1.7 An in situ End Notch Flexure (ENF) test

The specimen is placed under a **compressive load**, continuing until the crack propagates (again, detectable by a drop in the load-displacement trace). If propagation occurs at a load P_c , then G_{Ic} is given by equation (M10.1.3),

$$G_{IIc} = \frac{9P_c^2 a^2}{16E_f w^2 h^3} \quad (M10.1.3)$$

Unlike the Double Cantilever Beam (DCB) test, only one load cycle is required. As a general guide, Thermosets will exhibit stable mode-I growth and unstable mode-II growth, whereas the opposite is true for thermoplastics.

Learning Unit-2: M10.2

M10.2 Joining Of Composites

M10.2.1 Introduction

As in metal structures, local reinforcement is generally required where any hole (or cut-out) is placed in a structural part. Analogous to metal parts, reinforcement can be bonded or fastened to the structure. Local reinforcement may also be required in the vicinity of joints, either bonded or bolted and in locations where concentrated loads are introduced into the structure.

Joints often occur in transitions between major composite parts and a metal feature or fitting. In aircraft, such a situation is represented by **articulated fittings** on control surfaces as well as on

wing and tail components, which require the ability to pivot the element during various stages of operation. Tubular elements such as power shafting often use metal end fittings for connection to power sources or for articulation where changes in direction are needed. In addition, assembly of the structure from its constituent parts will involve either bonded or mechanically fastened joints or both.

Joints represent one of the greatest challenges in the design of structures in general and in composite structures in particular. The reason for this is that joints entail interruptions of the geometry of the structure and often, **material discontinuities**, which almost always produce local highly stressed areas, except for certain idealized types of adhesive joints such as scarf joints between similar materials.

One of the important factors affecting design of composite materials is the load carrying capability of the composite joints. The two commonly used types of load carrying joints, made of composite laminates, are:

- Mechanically fastened joints.
- Adhesive or bonded joints.

The classification of technological features as indicated in Figure M10.2.1 is conventional in appearance, but the difference lies in the content of joint manufacturing with due regard for the special properties of composite materials.

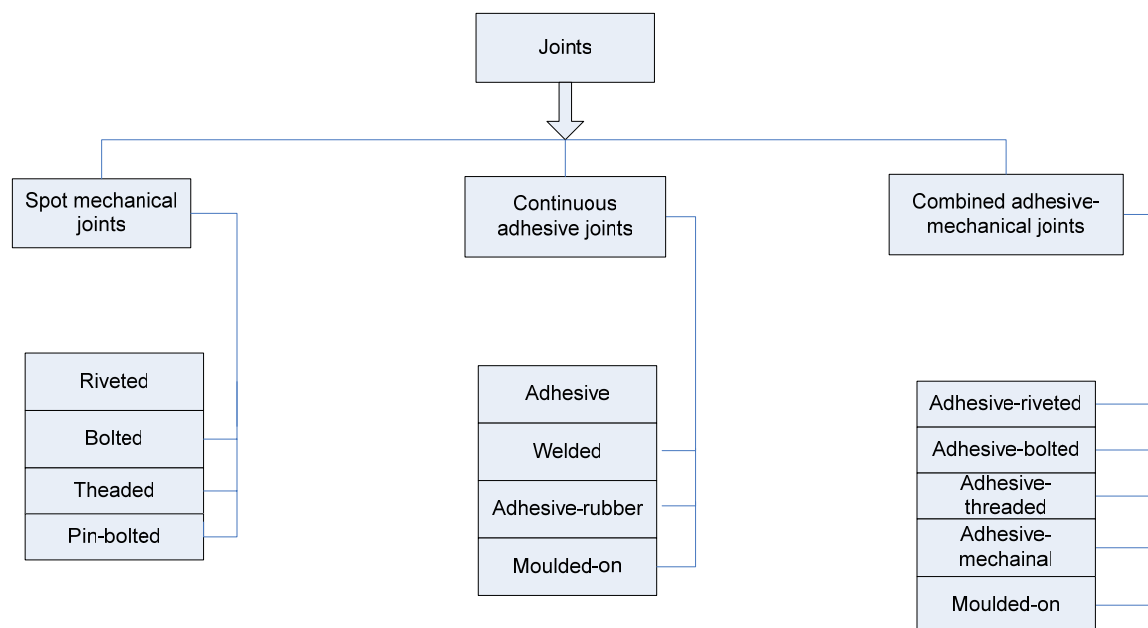


Figure M10.2.1 The classification of technological features

In principle, **adhesive joints** are structurally more efficient than mechanically fastened joints because they provide better opportunities for eliminating stress concentrations; Example: advantage can be taken of ductile response of the adhesive to reduce stress peaks. **Mechanically**

fastened joints tend to use the available material inefficiency. Sizeable regions exist where the material near the **fastener** is nearly unloaded, which must be compensated for by regions of high stress to achieve a particular required average load. In many cases, however, mechanically fastened joints cannot be avoided because of requirements for disassembly of the joint for replacement of damaged structure or to achieve access to underlying structure. Adhesive joints tend to lack **structural redundancy** and are highly sensitive to manufacturing deficiencies including poor bonding technique, poor fit of mating parts and sensitivity of the adhesive to temperature and environmental effects such as moisture. Assurance of bond quality has been a continuing problem in adhesive joints. While **non-destructive evaluation techniques** (ultrasonic and X-ray inspection) may reveal gaps in the bond, there is no present technique, which can guarantee that a bond, which appears to be intact does, in fact, have adequate load transfer capability. Thus mechanically fastened joints tend to be preferred over bonded construction in highly critical and safety related applications such as primary aircraft structural components, especially in large commercial transports, since assurance of the required level of structural integrity is easier to be guaranteed in mechanically fastened assemblies. As a rule, bonded joints prove to be more efficient for lightly loaded/non-flight critical aircraft structures whereas mechanically fastened joints are more efficient for highly loaded structures. Bonded construction tends to be more prevalent in smaller aircraft.

Figure M10.2.2 offers a list of the most common requirement of the joint design. It should be kept in mind that some of these requirements might become design variables in the course of the design process.

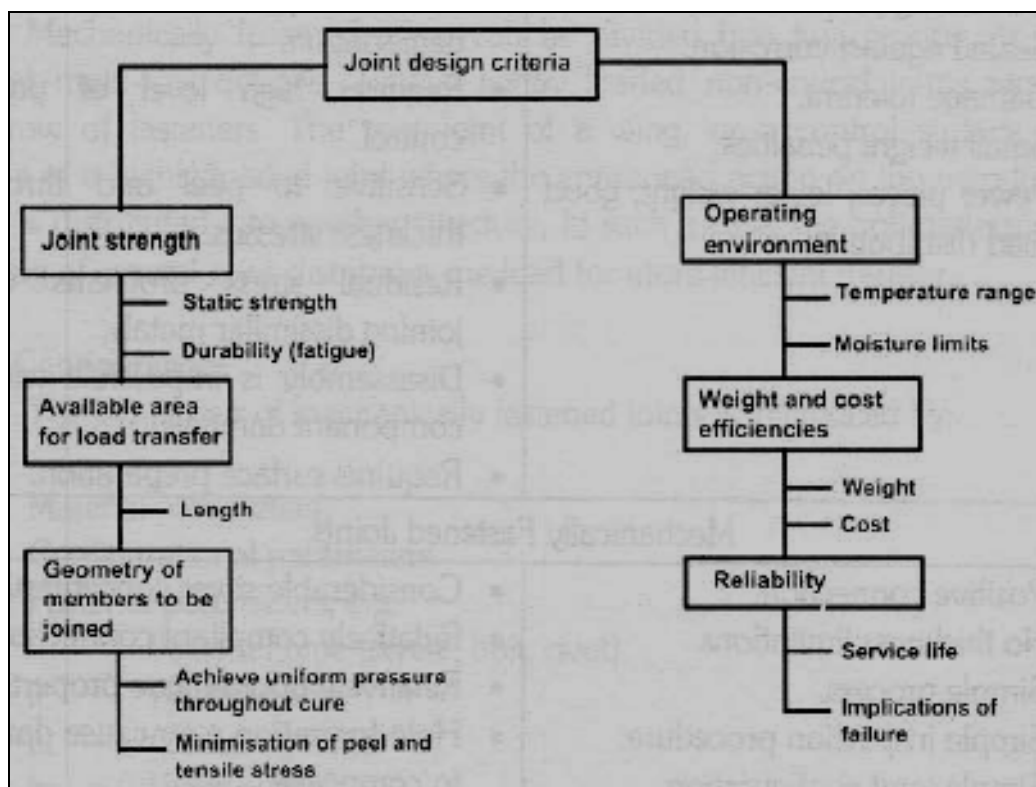


Figure M10.2.2 The most common requirement of the joint design

Geometry of the members being joined, for example, could be altered locally to facilitate joint design. **Reliability** includes an array of requirements, one of which is the implication of joint failure on system performance.

Type of joining to be used requires careful consideration of several parameters with a knowledge of the service that the joint is expected to provide. Load carrying joints usually have an overlap configuration.

Various advantages and disadvantages of bonded and mechanically fastened joints are as follows:

Advantages	Disadvantages
Bonded Joints	
<ul style="list-style-type: none"> • No stress concentration in adherents. • Stiff connection. • Excellent fatigue properties. • No fretting problems. • Sealed against corrosion. • Damage tolerant. • Small weight penalties. • Fewer pieces, lower weight, good load distribution. 	<ul style="list-style-type: none"> • Limits to thickness that can be joined with simple joint configuration. • Inspection difficulty. • Prone to environmental degradation. • Requires high level of process control. • Sensitive to peel and through-thickness stresses. • Residual stress problems when joining dissimilar metals. • Disassembly is impossible without component damage. • Requires surface preparation.

Table M10.2.1 Advantage and disadvantages of bonded fastened joints

M10.2.2 Mechanically Fastened Joints

The behaviour of composites in bolted joints differs considerably from that of metals. The brittle nature of composites necessitates more detailed analysis to quantify the level of various stress peaks. This is due to the fact that **stress concentrations** dictate part static strength to a larger extent than in metals. As a result, composite joint design is more sensitive to edge distances and hole spacings than metal joint designs.

Mechanically fastened joints can be divided into two groups, viz. single row and multi-row designs. Typical lightly loaded non-critical joints require a single row of fasteners. The root joint of a wing, or a control surface, is an example of a highly loaded joint where the entire load acting on the **aerodynamic surface** is distributed into another structure. In such a case, the bolt pattern design consisting of several rows distributes the load for more efficient transfer.

Advantages	Disadvantages
Mechanically Fastened Joints	
<ul style="list-style-type: none"> • Positive connection. • No thickness limitations. 	<ul style="list-style-type: none"> • Considerable stress concentration.

<ul style="list-style-type: none"> • Simple process. • Simple inspection procedure. • Simple joint configuration. • Not environmentally sensitive. • Provides through-thickness reinforcement and not sensitive to peel stresses. • No residual stress problems. • No surface preparation of component required. • Disassembly possible without component damage. • High tolerance to repeated loads. 	<ul style="list-style-type: none"> • Relatively compliant connection. • Relatively poor fatigue properties. • Hole formation may cause damage to composite. • Prone to fretting. • Prone to corrosion. • Large weight penalty.
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Table M10.2.2 Advantage and disadvantages of mechanically fastened joints

M10.2.2.1 Design Considerations

The behaviour of mechanically fastened joints is influenced by:

- a) Material parameters.
- b) Configurational parameters.
- c) Fastener parameters, e.g.
 - i. Fastener type (screw, bolt, rivet).
 - ii. Fastener size.
 - iii. Clamping force.
 - iv. Washer size.
 - v. Hole size.
 - vi. Tolerance.

The primary design considerations for bolted joints include joint strength, fastener type, local reinforcement, joint configuration, holes and pre-load: The process begins with the determination of a configuration for the joint. **Single lap joints** are normally adequate for thin laminates (up to about 5 mm in thickness). Fastener bending and initial bearing failure are primary areas of concern. **Double lap joints** are better for cyclic loads and generally stronger.

The use of mechanical fasteners to join composite structures is bound by certain constraints, which do not exist in the design of metallic joints. Care must be taken to select fasteners that are appropriate with the type of composite structures. Special types of fasteners are available for use on composites. These fasteners develop the full bearing capability of the composite without encountering **local failure modes** and are not susceptible to corrosion.

Fastener selection usually raises issues requiring decisions concerning laminate reinforcement, hole sizes and their location, drilling, fastener installation and inspection. Table M10.2.3 given below identifies various issues and proven design approaches to each issue. The Table indicates that the complexity of designing bolted joints arises from two primary sources, namely, (a)

composite laminates cannot re-distribute high local loads by yielding and plasticity; (b) composites are more easily damaged by drilling and fastener installation than metals.

Issue	Approach
Drilling damage.	<ul style="list-style-type: none"> • Closely controlled manufacturing operations. • Inspection of drilled holes.
High local stresses	<ul style="list-style-type: none"> • Larger fastener diameter. • Insert (bushing). • Increased laminate thickness (locally).
Preload relaxation	<ul style="list-style-type: none"> • Larger fastener head. • Washers (one or both sides). • Limit on installation torque.
Countersunk head	<ul style="list-style-type: none"> • Avoid, if possible. • Increased laminate thickness (locally).
Damage induced by installation of blind fasteners and drive rivets	<ul style="list-style-type: none"> • Specially designed blind rivets. • Verify joint strength with tests.

Table M10.2.3 The various issues and design approaches to each issue

Design of local reinforcement of the laminate to resist local stresses is an important step in the design of bolted joint. If reinforcement is required, a proven approach is to increase laminate thickness by addition of plies placed at $\pm 45^\circ$ and 90° to the primary load direction. A **quasi-isotropic laminate** provides the best bearing strength in any continuous fibre composite. Design of mechanically fastened joints has always been guided by the principle that the material being joined should fail before the fastener, and this is the practice with composites. The major structural limitation in designing mechanically fastened joints is the insufficient through-thickness strength of the laminates. This has given rise to the term pull-through strength.

Another area of concern is the bearing stress, which a fastener applies to the edge of the hole in a composite laminate as its axis rotates due to **secondary bending** of the joint. This condition can impose a severe limitation on a joint with limited stiffness. Further, composite's inability to support installation stresses of formed fasteners, such as solid rivets or blind fasteners poses another problem. In addition to **surface damage**, **sub-surface damage** to the laminate may occur. For this reason, use of these types of fastener is avoided.

M10.2.2.2 Failure Criteria

As in metallic joints, modes of failure in bolted joints of advanced composites are as follows:

- Tension or tearing failure** related to the net area through the fastener hole. The narrower the laminate, the more likely the chances of tensile failure.
- Shear out failure related to the shear areas emanating from the hole edge parallel to the load and determined by the end distance.
- Bearing failure** based on the projected area of the hole. Determined by the diameter of the hole. **Bearing strength** is greater than the **compressive strength** of the composite.
- Cleavage failure** is a mixed mode failure involving tension and bending.

The above failure modes are shown in Figure M10.2.3 and Figure M10.2.4. However, in practice, mixed modes of failure often occur. The allowable stresses in each of these modes are a function of the following:

- Geometry of the joint including the hole size, plate width and distance of the hole from the edge of the plate.
- The clamping area and pressure.
- The fibre orientations ply sequence.
- The moisture content and exposure temperature.
- The nature of stressing, e.g. tension or compression, sustained or cyclic and any out-of-plane loads causing bending.

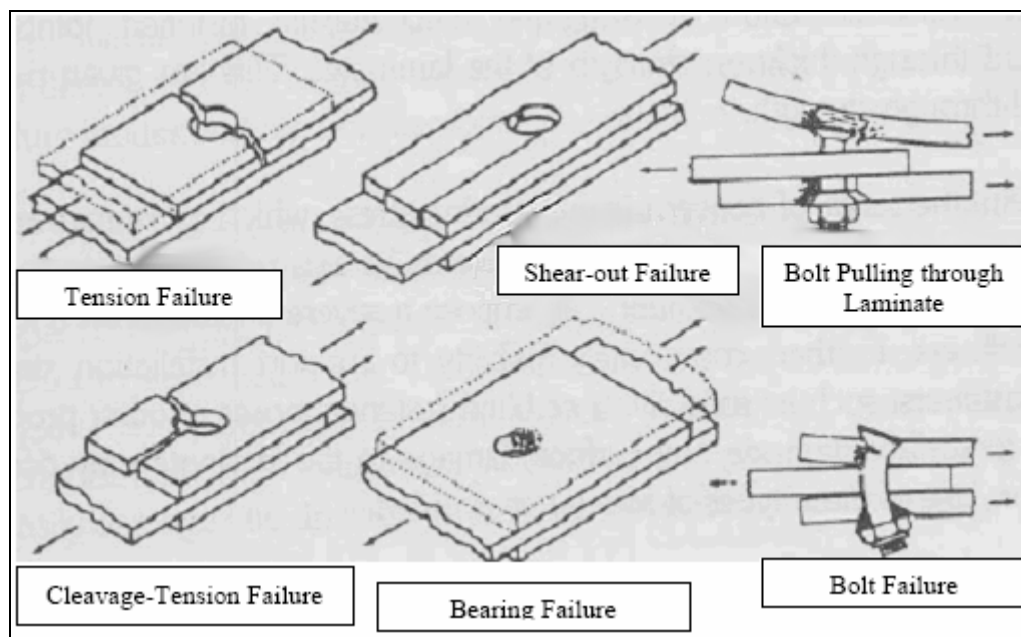


Figure M10.2.3 The failure modes

Accepted design practice is to select edge distances, plate thicknesses, and fastener diameters so that of all the probable failures would be the net section and the bearing. It is recommended that highly loaded structural joints be designed to fail in a bearing mode to avoid the **catastrophic failures** associated with net section failures.

The failure stresses will depend on the **degree of anisotropy** at the hole and hence on the local fibre orientation. Laminates containing a significant proportion of $\pm 45^\circ$ fibres have high shear strength and low stress concentrations at the hole. Therefore, they are relatively insensitive to edge distance.

M10.2.2.3 Fastener Selection

Fastener requirements for joining composite structures differ from those joining metallic structures. Fastener selection considerations for joining composites include corrosion

compatibility, fastener material, strength, stiffness, head configuration, importance of clamp-up, lightning protection, etc.

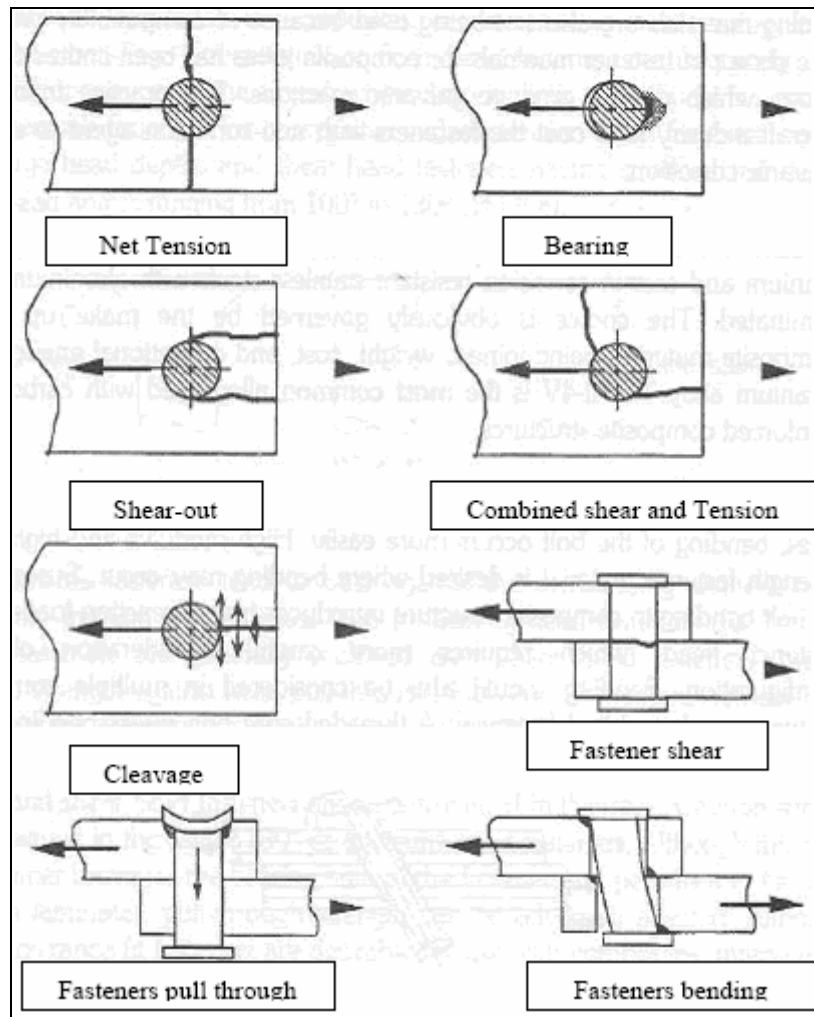


Figure M10.2.4 The failure modes

- Corrosion Compatibility:** Neither fibre glass nor aramid fibre reinforced composites cause corrosion problems when used with most fastener materials. Composites reinforced with carbon fibres are quite cathodic when used with materials such as aluminium or cadmium. Presence of galvanic corrosion between metallic fasteners and non-metallic composite laminates has eliminated several commonly used alloys from consideration. Conventional plating materials are also not being used because of compatibility problems. The choice of fastener materials for composite joints has been limited to those alloys, which do not produce galvanic reactions. The practice followed in aircraft industry is to coat the fasteners with anti-corrosion agent to alleviate galvanic corrosion.
- Fastener material:** The materials currently used in design include alloys of titanium and certain corrosion resistant stainless steels with aluminium being eliminated. The choice is obviously governed by the make up of the composite materials being joined, weight, cost,

and operational environment. Titanium alloy Ti-6Al-4V is the most common alloy used with carbon fibre reinforced composite structures.

- **Bolt bending:** Due to increased inter-laminar shear between the composite plies, bending of the bolt occurs more easily. High modulus and high tensile strength fastener material is desired where bending may occur. Susceptibility of bolt bending in composite structure introduces higher reaction loads on the fastener head, which requires more careful consideration of head configuration. Bending should also be considered in multiple component fasteners such as blind fasteners. A threaded core bolt resists bending much better than a smooth bore pull-type blind fastener as shown in Figure M10.2.5.

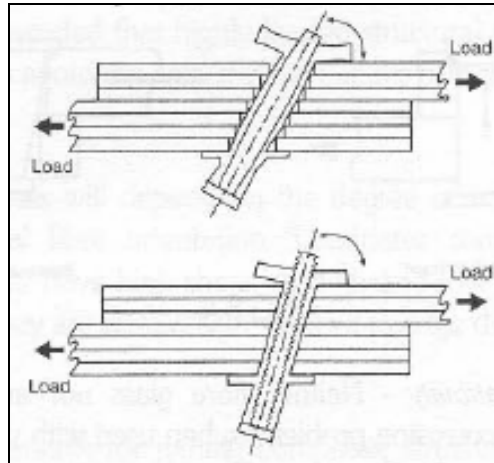


Figure M10.2.5 The Comparison of bending resistance between the threaded core bolt and the smooth bore pull-type blind fastener

- **Head configuration:** Composites are sensitive to high bearing loads than are metals. This means fastener heads should be designed with as much bearing surface area as practicable. The larger area improves pull-through and delamination resistance in composites, while reducing over-turning forces from bolt bending. **Countersunk or flush head fasteners** are frequently used on exterior surfaces of the aircraft where aerodynamic smoothness is required. Countersunk fasteners for composites include tension head fasteners having the large head depths and shear head fasteners having smaller head depths with head angles ranging from 100° to 130° as shown in Figure M10.2.6.

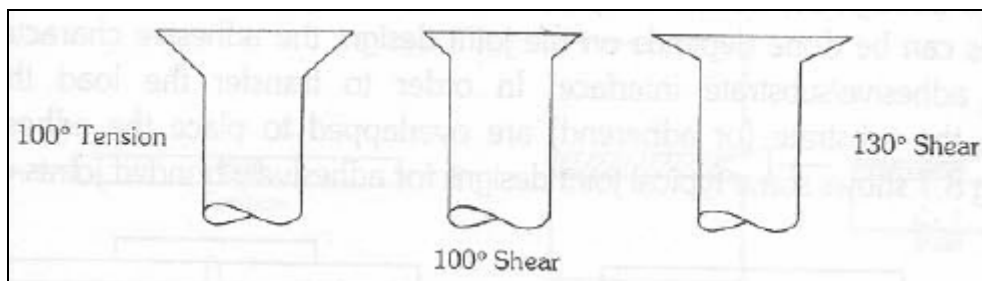


Figure M10.2.6 Countersunk fasteners for composites with different head depths

- **Countersunk fasteners:** tend to bear against the surrounding element more unevenly through the thickness than protruding head fasteners do. Tension head fasteners are generally preferred over shear head fasteners due to greater strength against head pull-through. However, if the joint element is so thin that the countersunk depth is greater than 70% of the element thickness, the tendency towards uneven bearing pressure in tension head fasteners is too great and shear head fasteners are recommended in this case. Caution should be observed in the use of 1300 countersunk head fasteners. Although this type of fastener increases the bearing area of the fastener and permits it to be used in thin laminates, pull-through strength can be adversely affected. Although; close tolerance fit fasteners are desirable for use with composites, **interference fit fasteners** cannot be used due to potential delamination of plies at the fastener hole.
- **Clamp up:** When tolerance fit holes are used, high clamp up appears to be beneficial for joint strength and fatigue life. The clamping forces, however, must be spread out over a sufficient area so that the compressive strength of the resin system is not exceeded and the composite crushed.

M10.2.3 Bonded Joints

As stated previously, adhesive joints are capable of high structural efficiency and constitute a resource for structural weight saving because of the potential for elimination of stress concentrations which cannot be achieved with mechanically fastened joints. However, due to lack of reliable inspection methods and a requirement for close dimensional tolerances in fabrication, aircraft designers have generally avoided bonded construction in primary structure.

In a structural adhesive joint, the load in one component must be transferred through the adhesive layer to another component. The efficiency with which this can be done depends on the joint design, the adhesive characteristics and the adhesive/substrate interface. In order to transfer the load through adhesive, the substrates (or adherend) are overlapped to place the adhesive in shear. Figure M10.2.7 shows some typical joint designs for adhesively bonded joints.

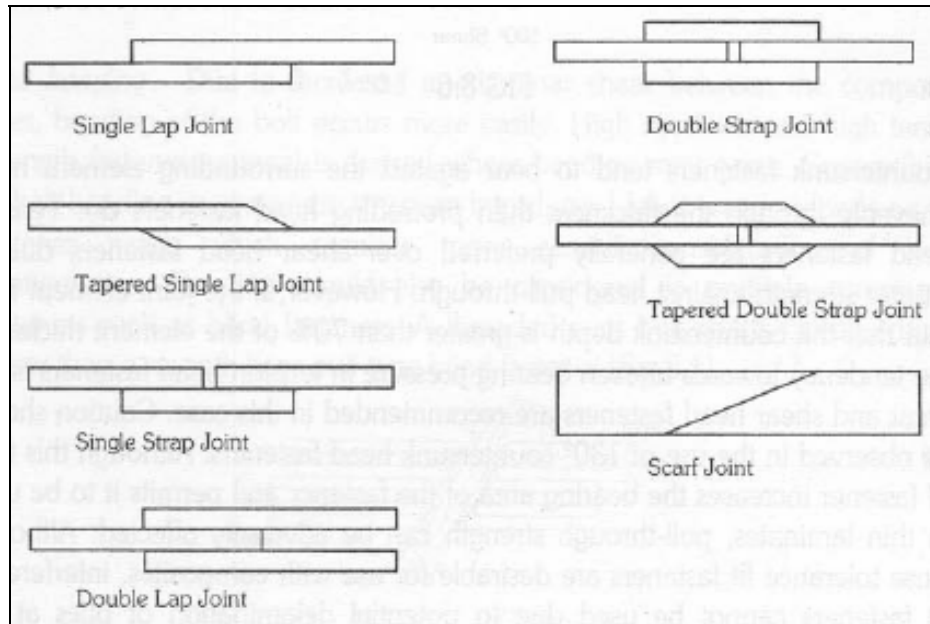


Figure M10.2.7 The typical joint designs for adhesively bonded joints

In general, adhesive joints are characterized by high **stress concentrations** in the adhesive layer. These originate, in the case of shear stresses, because of unequal axial straining of the adherends, and in case of peel stresses, because of **eccentricity** in the load path. Considerable ductility is associated with shear response of typical adhesives, which is beneficial in minimizing the effect of shear stress joint strength. The response of typical adhesives to peel stresses tends to be much more brittle than that to shear stresses. Reduction of peel stresses is desirable for achieving good joint performance.

The criteria for selecting an adhesive must be considered in view of the joint design. The joint design must ensure that the adhesive is loaded in shear as far as possible. Tension, cleavage and peel loading as shown in Figure M10.2.8 should be avoided when using adhesives.

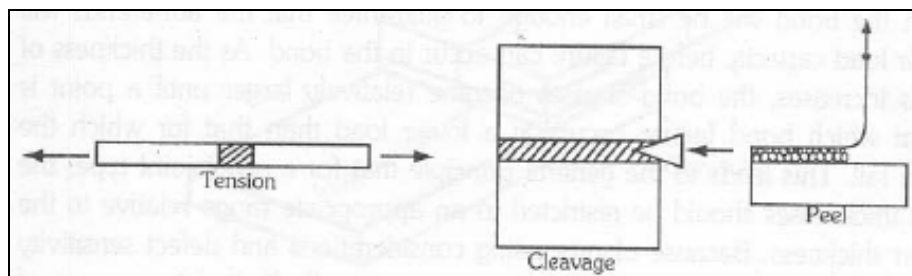


Figure M10.2.8 Tension, cleavage and peel loading in adhesives

M10.2.3.1 Design Considerations

The major considerations in the design of a bonded joint can be grouped into five categories. These include joint strength, environmental resistance, joint geometry, selection of the adhesive system and processing. The first step in the process of adhesive joint is to determine a dimensional configuration, which minimizes tensile and peel stresses. Once this is accomplished,

the next task is to select an adhesive system, which best satisfies static strength, fatigue life and environmental requirements. The third step is the development of process specifications for the joint to include details for surface preparation, curing the joint and maintaining pressure during cure, if necessary. The joint strength is typically verified analytically or by structural tests or both. Bonded joint strength vis-a-vis adherends thickness different types of joints is represented in Figure M10.2.9.

From the standpoint of joint reliability, it is vital to avoid adhesive layer to be the weak link in the joint. This means that whenever possible, the joint should be designed to ensure that the adherends fail before the bond layer. This is because failure in the adherends is fibre controlled, while failure in the adhesive is resin dominated and thus subject effects of **voids** and other defects, thickness variations, environmental effects, processing variations, deficiencies in surface preparation and other factors that are not always adequately controlled. This is a significant challenge since adhesives are inherently much weaker than the composite or metallic elements being joined. However, the objective can be accomplished by recognizing the limitations of the joint geometry being considered and placing appropriate restrictions on the thickness dimensions of the joint for the each geometry. In each type of joint, the adherend thickness may be increased as an approach to achieve higher load capacity. When the adherends are relatively thin, results of stress analysis show that for all types of joints, the stresses in the bond will be small enough to guarantee that the adherends will reach their load capacity before failure can occur in the bond. As the thickness of adherends increases, the bond stresses become relatively larger until a point is reached at which bond failure occurs at a lower load than that for which the adherends fail. This leads to the general principle that for a given joint type, the adherend thicknesses should be restricted to an appropriate range relative to the bond layer thickness. Because of processing considerations and defect sensitivity of the bond material, bond layer thicknesses are generally limited to a range of 0.125 to 0.40 mm.

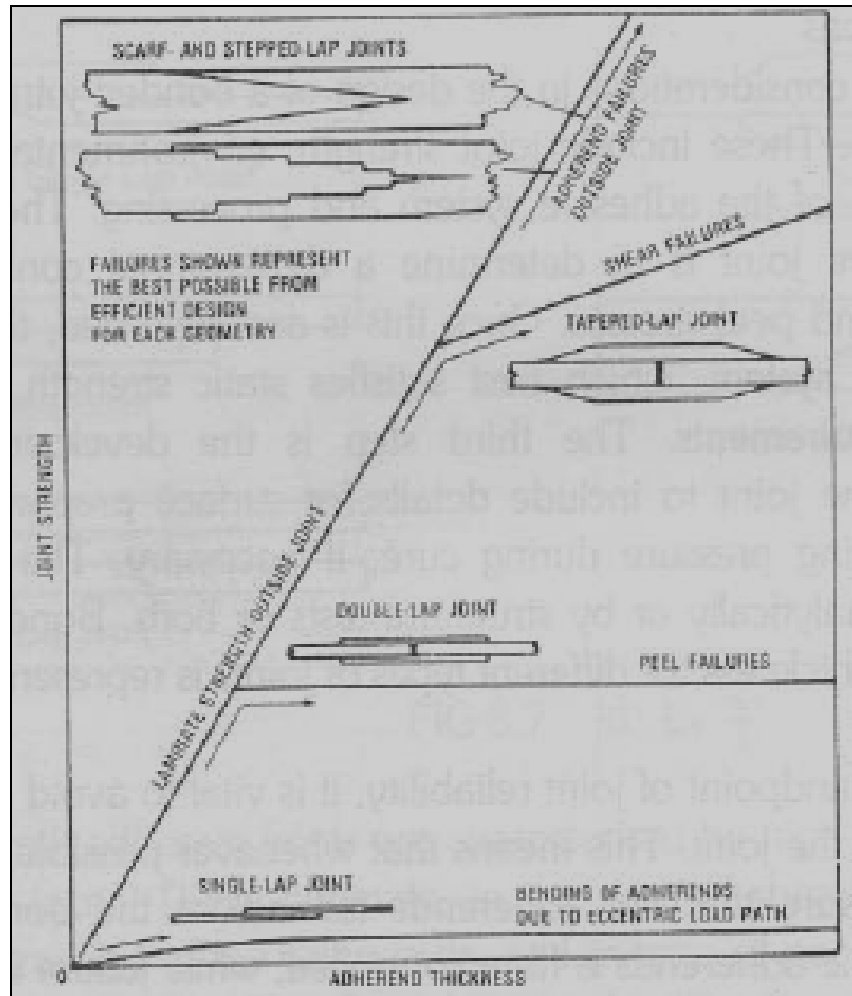


Figure M10.2.9 Bonded joint strength Vs adherends thickness for different types of joints

M10.2.3.2 Failure Criteria

A number of failure modes may occur in bonded composite joints because of their anisotropic nature. In the adherends, failure can be tensile, inter-laminar or transverse. There may be cohesive failure also, which can occur in the adhesive. Various failure modes are shown in Figure M10.2.10 and Figure M10.2.11.

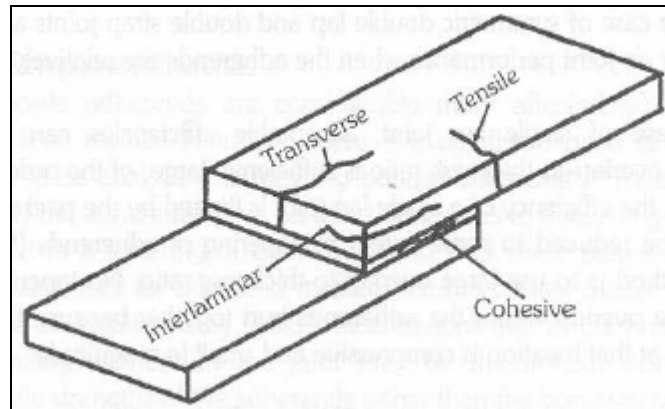


Figure M10.2.10 The failure modes

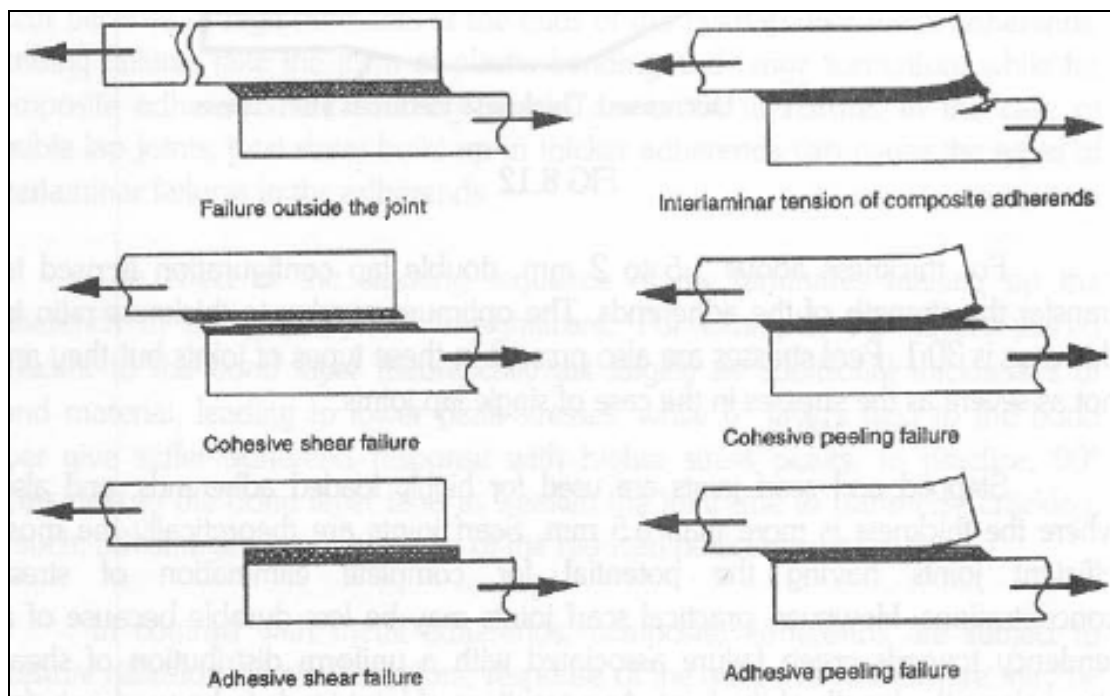


Figure M10.2.11 The failure modes

M10.2.3.3 Effect of Joint Geometry

Single and double lap joints with uniformly thick adherends are the least efficient joints. These joints are suitable primarily for thin structures with low running loads i.e. load per unit width. Of these, single lap joints are the least capable because the eccentricity of this type of geometry generates significant bending of the adherends that magnifies the **peel stresses**. Peel stresses are also present in the case of **symmetric double lap** and double strap joints and become a limiting factor on joint performance when the adherends are relatively thick.

In case of single lap joint, acceptable efficiencies can be achieved provided the overlap-to-thickness ratio is sufficiently large, of the order of 50:1. As stated above, the efficiency of a single lap joint is limited by the peel stresses. Peel stresses can be reduced to some extent by tapering of adherends (as shown in Figure M10.2.12) but the main method is to use large

overlap-to-thickness ratio. No tapering is needed at ends of the overlap where the adherends butt together because the transverse normal stresses at that location is compressive and small in magnitude.

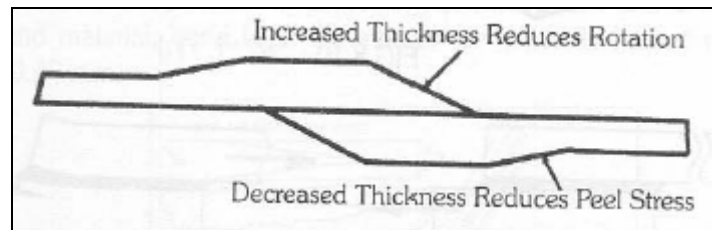


Figure M10.2.12 Reduction of the peel stresses by overlapping of tapered adherends

For thickness above 1.5 to 2 mm, double lap configuration is used to transfer the strength of the adherends. The optimum overlap-to-thickness ratio in this case is 30: 1. Peel stresses are also present in these types of joints but they are not as severe as the stresses in the case of single lap joints.

Stepped and scarf joints are used for highly loaded adherends, and also where the thickness is more than 6.5 mm. Scarf joints are theoretically the most efficient joints having the potential for complete elimination of **stress concentrations**. However, practical scarf joints may be less durable because of a tendency towards **creep failure** associated with a uniform distribution of **shear stress** along the length of the joint. As a result, scarf joint tends to be used only for repairs of very thin structures. Stepped joints represent a practical solution of bonding thick members. Stepped joints have been extensively used where adherends are subjected to high load intensities. High loads can be transferred if number of short steps of small rise (thickness increment) in each step is used, while maintaining sufficient overall length of the joint. Factors influencing the joining strength are number of steps, length and thickness of each step.

M10.2.3.4 Behaviour of Composite Adherends

Composite adherends are considerably more affected by inter-laminar shear stresses than metals. Therefore, there is a significant need to account for such effects in **stress analysis** of adhesively bonded composites. Transverse shear deformations of the adherends have an effect analogous to thickening of the bond layer and result in a lowering of both shear and peel stress peaks. In addition, because the resins used for adherend matrices tend to be less ductile than typical adhesives, and are weakened by **stress concentrations** due to the presence of the fibres, the limiting element in the joint may be the inter-laminar shear and transverse tensile strengths of the adherends rather than the bond strength.

In the case of **single lap joints**, bending failures of the adherends may occur because of high moments at the ends of the overlap. For metal adherends, bending failures take the form of plastic bending and hinge formation, while for composite adherends the bending failures are brittle in nature. In the case of **double lap joints**, **peel stress** build up in thicker adherends can cause the types of inter-laminar failures in the adherends.

The effect of the stacking sequence of the laminates making up the adherends in composite joints is significant. For example, 900 layers placed adjacent to the bond layer theoretically act largely as additional thicknesses of bond material, leading to lower peak stresses, while layers next to the bond layer give stiffer adherend response with higher stress peaks. In practice, 900 layers next to the bond layer tend to weaken the joint due to transverse cracking, as such; advantage cannot be taken of the reduced peak stresses.

In contrast with metal adherends, composite adherends are subject to **moisture diffusion effects**. Therefore, response of the adhesive to moisture may be more significant issue for composite joints.

M10.2.3.5 Effects of Bond Defects

Defects in adhesive joints, which are of concern, include **surface preparation deficiencies**, **voids** and **porosity**, and thickness variations in the bond layer. Of the various defects, which are of interest, surface preparation deficiencies are probably of greatest concern. These are particularly troublesome because there are no current **non-destructive evaluation techniques**, which can detect low interfacial strength between the bond and the adherends.

For joints, which are designed to ensure that the adherends are the critical elements, tolerance to the presence of porosity and other types of defect is considerable. **Porosity** is usually associated with over-thickened areas of the bond, which tend to occur away from the edges of the joint where most of the load transfer takes place. Therefore, it is a relatively **benign effect**, especially if peel stresses are minimized by adherend tapering. If peel stresses are significant, as in the case of over-thick adherends, porosity may grow catastrophically and lead to non-damage-tolerant joint performance. In the case of bond thickness variations, these usually take place in the form of thinning due to excess resin bleed at the joint edges, leading to overstressing of the adhesive in the vicinity of the edges. Inside tapering of the adherends at the joint edges can be used to compensate for this condition. Bond thicknesses should be limited to ranges of 0.12-0.24 mm to prevent significant porosity from developing. Common practice involves the use of film adhesives containing scrim cloth, some forms of which help to maintain bond thicknesses. It is also common practice to use mat carriers of chopped fibres to prevent a direct path for access by moisture to the interior of the bond.

M10.2.3.6 Surface Pre-Treatment Prior to Bonding

Surface pre-treatment requires removal of contaminants such as oils, mold lubricants or general dirt. Techniques used for surface pre-treatment are:

1. **Peel ply method** in which one ply of fabric should be installed at the bonding surface and removed just prior to bonding thereby exposing the clean bondable surface. In this technique, a closely woven nylon or polyester cloth is used as the outer layer of the composite during lay-up. This ply is torn or peeled away just before bonding. The basic idea is that the tearing or peeling process fractures the resin-matrix coating and exposes a clean, virgin roughened surface for the bonding process.

2. **Abrasion and solvent cleaning** to remove abrasion products followed by a solvent wipe. Abrasion increases the surface energy of the surfaces to be bonded and removes any residual contamination. The abrading operation should be conducted with care to avoid exposing or rupturing of the reinforcing fibres from the surface.

A typical cleaning sequence would be to remove the peel ply and then lightly abrade the surface with a dry grit blast. After **grit blasting**, any remaining residue on the surface may be removed by **dry vacuuming or wiping** with a clean, **dry cheese cloth**.

M10.2.3.7 Joint Manufacture

Bonded joints can be made by gluing together pre-cured laminates with a suitable adhesive. Alternatively, bonded joints can be made by forming joints during the manufacturing process in which the joint and the laminate are cured at the same time (**co-cured**).

M10.2.3.8 Adhesive Selection

The selection of adhesives is based on the strength requirements over the expected service temperature range and the type of equipment available for bonding. Different types of adhesives are available which provide different ranges of adhesive bonding shear and peel strengths at various service temperatures. Adhesives for structural bonding can be categorized into three main physical forms in which they are used - (a) **films**, (b) **pastes** and (c) **foams**. Although films are easier to handle and provide a more uniform bond line thickness than paste adhesives, lack of refrigerated storage equipment sometimes necessitates use of paste adhesives. Foam adhesives are used for stabilizing and splicing pieces of **honeycomb core**. The criteria for selection of adhesive are as follows:

- The adhesive must be compatible with the adherends and able to retain its required strength when exposed to in-service stresses and environment.
- The joint should be designed to ensure failure in one of the adherends rather than failure within the adhesive bond line.
- **Thermal expansion** of dissimilar materials must be considered. Due to large thermal expansion difference between graphite composite and aluminium, adhesively bonded joints between these two materials are likely to fail during cool down from elevated temperature cures as a result of the **thermal stresses** induced by their differential expansion coefficients.
- Proper joint design should be used avoiding tension, peel or cleavage loading. If peel forces cannot be avoided, a lower modulus adhesive having high peel strength should be used.
- Surface preparation should be conducted carefully, avoiding contamination of the bond line with moisture, oil, etc.
- The adhesive should be stored at the recommended temperature.
- Use of adhesives that evolve volatiles during cure should be avoided.

- The recommended pressure and proper alignment fixtures should be used. The bonding pressure should be great enough to ensure that the adherends are in intimate contact with each other.
- **Traveller coupons** should always be made for testing.
- The exposed edge of the bond joint should be protected with an appropriate sealing compound.

The major advantages of film adhesives are that they are easier to apply and do not require mixing equipment. They have more uniform viscosity and composition and provide more bond line thickness uniformity in a joint than do paste adhesives. The major disadvantages of film adhesives are that refrigeration is required for storage. In addition, film adhesives are more expensive than pastes and require heat and pressure to achieve satisfactory bonds.

Paste adhesives have a long shelf life and do not require refrigeration. However, they have to be mixed before application, which introduces possible human error of incomplete mixing or improper weighing. Further, paste adhesives have lower strength properties than film adhesives especially for elevated temperature service.

Foam adhesives are used in honeycomb repair to fill gaps in splice areas or between edge members and honeycomb core. In addition, they are used to fill voids and eliminate moisture paths through splice areas.

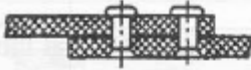
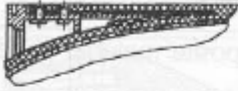
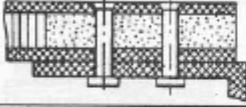
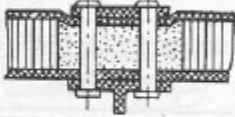
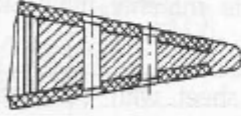
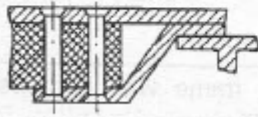
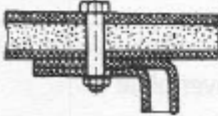
M10.2.4 Test Verification

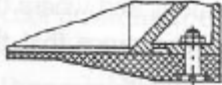
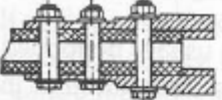









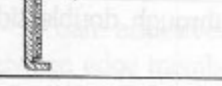

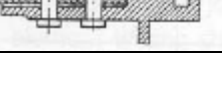
In addition to joint coupon testing, which is performed to obtain baseline data, element testing should be performed to verify joint analysis, failure mode, and location. This is particularly important for primary connections and where the load transfer is complex. The purpose of testing is to obtain assurance that the joint behaves in the predicted manner or where analysis is inadequate.

The bolted joint element or sub-component tests are usually performed at ambient conditions to fully characterize load transfer details. Tests at other than ambient conditions are necessary in cases where the low or elevated temperatures with associated moisture contents substantially change the load distributions.

M10.2.5 Typical Joint Designs

A variety of mechanical, adhesive-mechanical and combined joints are shown in the following Table M10.2.3.

<i>Joint Design</i>	<i>Structural and Technological Characteristics</i>
	Standard composite material one-shear joint
	Frame joint with composite material shell
	Three-ply carbon-filled plastic panel joint with a spar
	Joint of two honeycomb-filled panels with each other and the rib
	Joint of wedge-shaped double sided countersink joint stacks
	Joint of fairing with the frame through double sided countersink riveting
	Comb-joint assembly with a three-ply carbon-filled plastic panel

	Comb-joint assembly with a three-ply carbon-filled plastic panel
	Bracket joint with a three-layer panel with triangular filler
	Composite material blade joint with an attachment
	Composite material cylindrical shell joint with a metal frame
	Three-ply skin joint with a rib and a spar
	Local hardening of metal structures to increase their rigidity in predetermined direction
	Stepped joint of three-ply composite material panels with metal
	Joint of stringer with skin
	Joint of aft section with helicopter rotor spar
	Adhesive joint of a frame with a composite material shell
	Stepped lap joint of composite material main skin with metal
	Mechanical joining of metal sheet with composite material section
	Adhesive riveted joint of a frame with composite material skin and its outer sealing
	Adhesive riveted joint of a profiled section and composite material skin with a cover plate

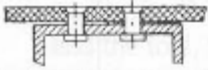
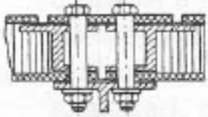
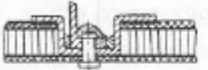
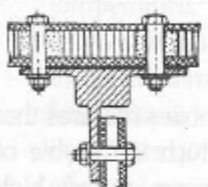
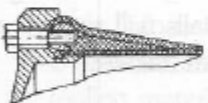
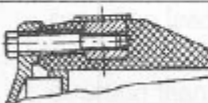
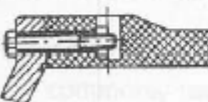


	Adhesive riveted joint of a profiled section with a composite material skin
	Three-ply composite material joined by metal inserts, cover plate and a supporting member
	Joint of three-ply stepped edge panel with intermediate support
	Joint of three carbon filled plastic panels with a spar
	Flanged bolted joint
	Flanged pin-bolted joint with a stop
	Standard design of composite material flanged bolted joint with metal
	Flanged bolted joint with an intermediate frame; the joint is reinforced by fibre glass circular layers
	Pin-studded joint with an intermediate frame; the joint is reinforced by fibre glass circular layers

Table M10.2.4 A variety of mechanical, adhesive-mechanical and combined joints

Learning Unit-3: M10.3

M10.3 Environmental Effects on Composites

Composite usage has increased enormously mainly due to the advantages of lightweight, specific strength and stiffness, **dimensional stability**, **tailor-ability of properties** such as coefficient of thermal expansion and high thermal conductivity. Environmental effects on these properties may compromise a structure and must be considered during the design process.

This module deals with the major environmental concerns for the composite designer, problems encountered with these environments in the past and some materials or protective systems effectively used.

Different environmental factors along with their effect on composites are briefly discussed in the subsequent paras.

M10.3.1 Biological Attack

Biological attack on composite materials may consist of fungal growth or marine fouling. Fungal growth does not appear to be as damaging as the wet conditions that promote growth. Fungicide has been mixed in with resins to retard this growth. Even though marine organisms will grow on composite surfaces, mechanical properties do not appear to be affected and the fouling can be removed by scraping. Composites with graphite fibres have been used in medical applications for both internal and external purposes. **Internal composite structures** such as artificial joints or plates for bone fracture support must be bio-compatible or the material may degrade over time. **External composite** designs (such as artificial limbs or orthotic braces) may experience **impact damage**, flexural and torsional loading during use.

M10.3.2 Fatigue

Fatigue, either through mechanical loads or acoustic vibrations, can cause crack growth or local defect formation. Fatigue design depends not only on the load but also on the use temperature range and amount of moisture present. Very cold temperatures (below -50°C) may increase the stiffness of some composite materials thereby increasing the susceptibility to fatigue damage.

M10.3.3 Fluids

M10.3.3.1 (a).Moisture

Moisture is present in the operational environment in which a composite is manufactured and throughout its useful life. Water acts as a plasticiser when absorbed by the matrix, softening the material and reducing some properties of the laminate. Moisture may also migrate along the fibre-matrix interface thereby affecting the adhesion. Moisture in composites reduces matrix dominated properties such as **transverse strength, fracture toughness** and **impact resistance**. Lowering of the glass transition temperature may also occur in epoxy and polyimide resins with an increase in absorbed moisture (as shown in Figure M10.3.1). **Debonding** can occur due to formation of discontinuous bubbles and cracking in the matrix. Mechanical properties can be reduced even further if heat is present or if the composite is **under-cured** or has a large amount of voids.

Moisture is absorbed into the composite until a saturation point is reached. This has been described as a **non-Fickian process**, meaning the rate of relaxation in the material due to water absorption is comparable to the diffusion rate of water. As the material properties change, such as decrease in glass **transition temperature**, the **diffusion process** changes. The mechanical properties degrade in relation to the amount of moisture absorbed, with no further deterioration after saturation is reached. Strength reductions in polyester laminates have been found to be 10-15% while epoxy resins are less vulnerable.

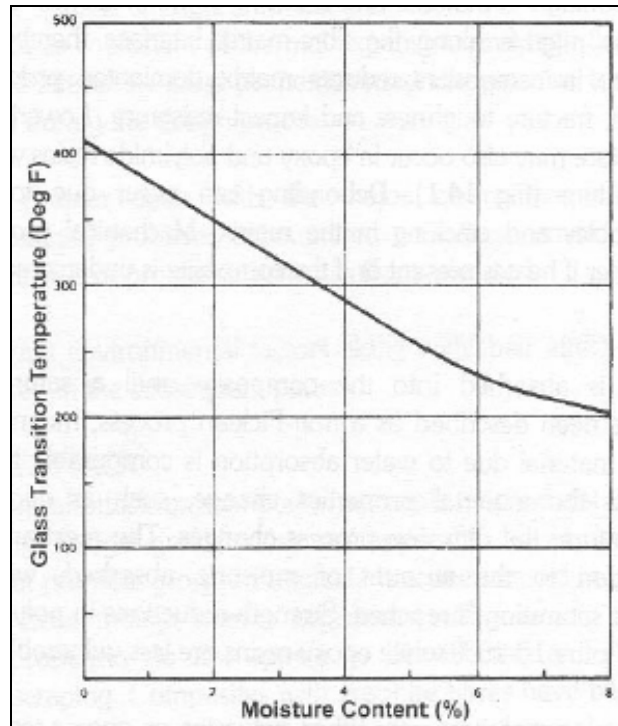


Figure M10.3.1 The relation between Moisture content and Glass transition temperature

Fibre glass composites with either polyester or epoxy resins have been used extensively in marine structural applications due to their strength to weight characteristics and resistance to the marine environment. Glass reinforcement is preferred over carbon fibres due to carbon's electrical conductivity, which may result in severe dissimilar metals galvanic corrosion with sea water acting as an electrolyte. This is because carbon along with metallic alloys is in the electromotive series of alloys commonly used in aircraft structures. A galvanic cell can thus be formed in the presence of moisture or any other electrolyte between carbon and contacting metal. Carbon, which is the cathodic end of the series and act as a noble metal, is impervious to corrosion itself but will accelerate corrosion in the adjacent less noble metal. Special corrosion control techniques are employed when CFRP components are placed in contact with aluminium components in aircraft assemblies. A fibre glass/epoxy ply is laid up and cocured with the carbon/epoxy plies. A faying surface sealant is applied between the two components. Aluminium parts are anodized, primed and painted prior to assembly (as shown in Figure M10.3.2).

M10.3.3.1 (b) Aircraft fluids

The aircraft fluid environment consists of fuel, hydraulic fluid, lubricants, de-icing compounds and water. Polysulphone has been found to be sensitive to phosphate ester based hydraulic fluids. Some polymer resins such as PEEK may have lower glass **transition temperatures** after exposure to fluids with a high aromatic content. The fuel-water immersion appeared to be the most damaging, reducing the tensile strength of graphite/epoxy and Kevlar composites by 11 % and 25 % respectively.

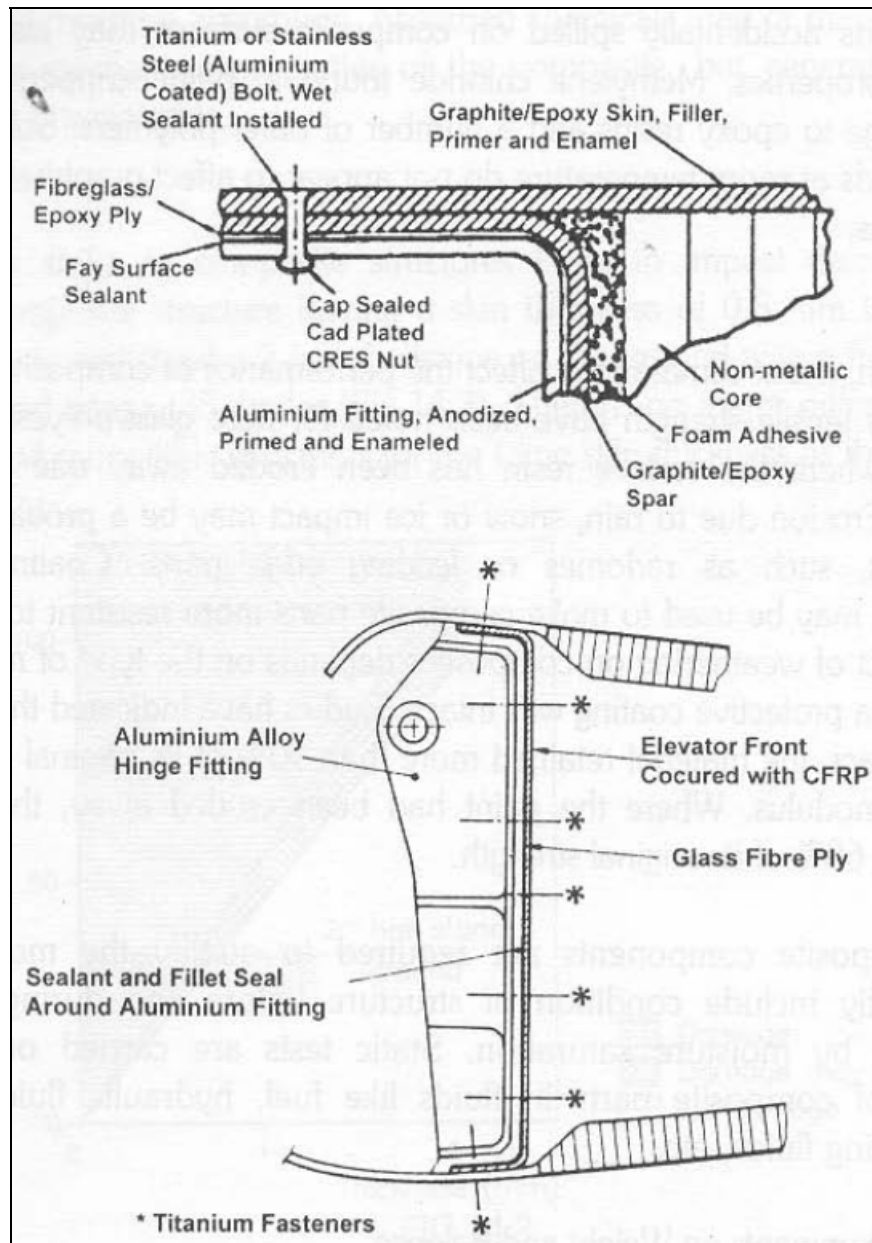


Figure M10.3.2 Aluminium parts anodized, primed and painted prior to assembly

M10.3.3.1 (c) Automotive Fluids

The automotive fluid environment consists of gasoline, oil, battery acid, brake fluid, transmission fluid and coolant. Most of the composites in a moist high temperature (150°) environment exhibited micro-cracking. The amount of moisture absorbed, as measured by weight gain, is directly related to the change in mechanical properties. Salt water, antifreeze and gasoline produce most pronounced effects on composites.

M10.3.3.1 (d) Other Fluids

Liquids accidentally spilled on composite surfaces may also affect the mechanical properties. Methylene chloride found in paint strippers may cause severe damage to epoxy resins and a number of other polymers. Solvents, bases and weak acids at room temperature do not appear to affect graphite/epoxies and **Kevlar/epoxies**.

M10.3.3.2 (a) Weathering

Warm, moist climate may affect the performance of composites. Decrease of 10-20% in tensile strength has been noted for fibre glass/polyester and fibre glass/epoxy where the surface resin has been eroded away due to extended **weathering**. Erosion due to rain, snow or ice impact may be a problem for some aircraft parts, such as radomes or leading edge parts. Coatings such as polyurethane may be used to make composite parts more resistant to this type of erosion. Effect of weathering on composites depends on the type of material used and whether a protective coating was intact. Studies have indicated that where the paint was intact, the material retained more than 90% of its original strength and 80-90% of modulus. Where the paint had been eroded away, the composite retained only 68% of its original strength.

Composite components are required to qualify the moisture tests, which broadly include condition of structure before and during static and **fatigue tests** by moisture saturation. **Static tests** are carried out following immersion of composite parts in fluids like fuel, hydraulic fluids, cleaning agents, de-icing fluids, etc.

M10.3.3.2 (b) Effect of Contaminants on Weight and Balance

Fluids absorbed by or otherwise introduced into a structure induce weight gains and may cause out-of-balance conditions in flight control surfaces. Contamination detected should always be evacuated, the leakage paths identified, repaired and the structure re-sealed.

M10.3.3.2 (c) Effects of Contaminants on Structural Integrity

Dimensional swelling of the resin matrix generally results from exposure to high humidity at high temperatures, exposure to many aircraft fluids, to chemical paint strippers and to a variety of common solvents. Absorbed moisture lowers the glass transition temperature of a laminate and may be conducive to additional micro cracking within the matrix, which in turn increases the potential for additional moisture absorption. Absorbed chemicals may or may not affect the structural or mechanical properties of the composite, but generally render the affected part un-repairable.

M10.3.4 Hail

Hail strike to composite structures leads to **impact damage**. For this purpose, composite structure having a skin thickness of 0.8 mm is protected at design stage to withstand a 2-inch hailstone on the ground with a free fall velocity of 33 m/s and energy 35 Joules (Figure M10.3.3). There is no major difference between **sandwich** and **monolithic structures** for the same skin thickness as they are equally resistant.

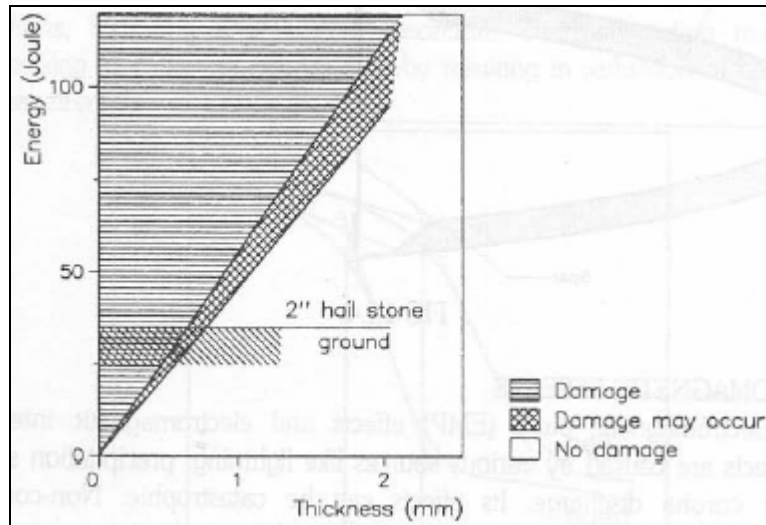


Figure M10.3.3 Impact dallage due to Hail strike in composite structures

M10.3.5 Foreign Object Damage

This type of damage is caused due to foreign objects striking the surface of composites causing possible **localised damage** or **delamination**, etc. It includes ballistic damage, damage from sand, dust, stones and more often from bird strike. The impact resistant of composite materials can be controlled by the choice of reinforcement and matrix. The matrix can be altered by addition of plasticisers, which increase the strain to failure. In addition, radomes and leading edges are designed to protect the structural parts from **bird impact damage**. Some of these protections are shown in Figure M10.3.4.

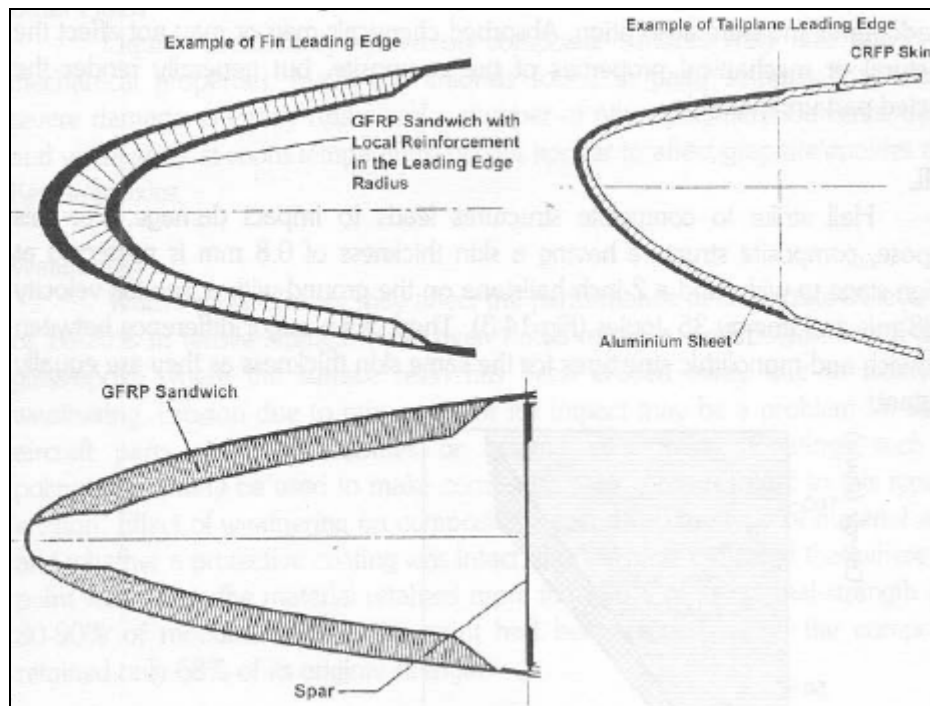


Figure M10.3.4 The protections of surface of Composite materials by design of radomes and leading edges

M10.3.6 Electro Magnetic Effects

Electromagnetic pulse (EMP) effects and **electromagnetic interference (EMI) effects** are caused by various sources like **lightning**, **precipitation static (p-static)** or **corona discharge**. Its effects can be **catastrophic**. Non-conductive composites provide little shielding effectiveness, while conductive composites like carbon/epoxy provide varying degree of protection. Fuel tanks, electrical equipment, etc. require isolation from static discharge. Lightning protection schemes can sometimes serve the dual purpose of providing lightning protection and p-static protection.

The objective is to bleed off the static charge prior to any significant build-up that could cause a fire or explosion or precipitate electromagnetic interference with the on-board electrical equipment. The shielding effectiveness of composites can be improved by metal coatings.

M10.3.7 Temperature Effects

Temperature effects on composite materials include **cryogenic temperatures**, **elevated temperatures** and **thermal cycling** between these extremes. Cryogenic temperatures do not appear to affect the mechanical properties of graphite/epoxies or graphite/polyimides significantly. Elevated temperatures for a prolonged period of time can seriously affect the properties of a composite, with even greater effect if moisture is present. Loss of stiffness with temperature and **ageing** is indicated in Figure M10.3.5. Susceptibility to **matrix softening** is not only dependent on the resin but also the lay-up. Temperature effects are not limited to the matrix materials. Extended operation at 350°C (660°F) and 450°C (840°F) can cause oxidation of low modulus **PAN-based fibres** and high modulus **PAN-** or **Pitch-based fibres**, respectively. Oxidation resistance can be improved with higher purity fibres. Thermal cycling conditions are common for a number of applications, including aircraft and spacecraft. Thermal cycling may induce **micro-cracking** in some composites thereby resulting in reduction of compressive and shear strength.

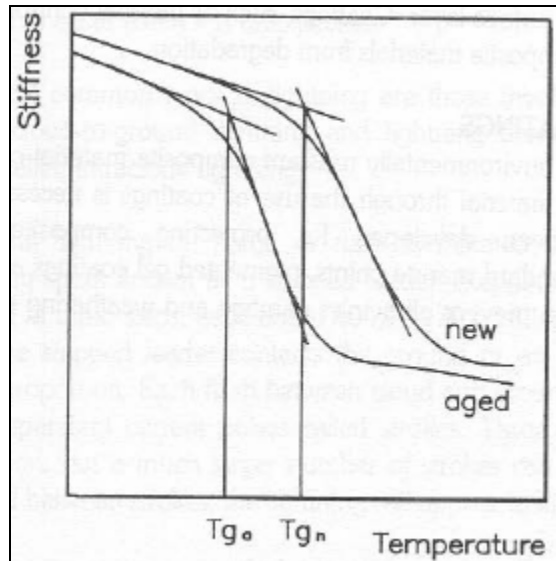


Figure M10.3.5 The effect of temperature and ageing on stiffness of composite Materials

Protection against temperature effects can be achieved at the design stage itself by:

- Selection of resin system with high **glass transition temperature**.
- Potential **degradation** taken into account in the analysis and fatigue test.
- Protection against moisture exposure.

M10.3.8 Overheat Conditions

Heat generated by lightning strikes has been known to vaporize matrix resins and create large areas of delamination and fibre fracturing on **composite rudders, ailerons, wing and stabilizer tips, nose domes and nacelle cowling**. When exposed to hot gases over long periods, polymeric resin binders can become completely destroyed through a process of thermo-oxidation. Preventive methods may consist of application of heat resistant ablative coatings.

M10.3.9 Effect of Ultra Violet Radiation

Ultraviolet radiation is a band of light from 300 to 4000 Å. Ultra-violet radiation may cause degradation through molecular weight change and cross-linking in the resin system. However, this damage is generally limited to darkening of the resin in the surface layer. Coatings, such as thermal control tape, have been used to protect composite materials from degradation.

M10.3.10 Protective Coatings

When an environmentally resistant composite material cannot be utilized, protection of the material through the use of coatings is necessary. A variety of coatings have been developed for protecting composites from various environments. Standard marine paints, pigmented gel coatings and polyurethanes have been used to prevent ultraviolet damage and weathering erosion of marine composites.

M10.3.11 Hygrothermal Stresses and Strains in a Lamina

Composite materials are generally processed at high temperature and then cooled down to room temperatures. For polymeric matrix composites, this temperature difference is in the range of 200 to 300° C, while for ceramic matrix composites, it may be as high as 1000° C. Due to mismatch of the coefficients of thermal expansion of the fibre and matrix, residual stresses result in a lamina when it is cooled down. Also, it induces expansional strains in the lamina. In addition, most polymeric matrix composites can absorb or deabsorb moisture. This moisture change leads to swelling strains and stresses similar to those due to thermal expansion. Laminate where lamina are placed at different angles have residual stresses in each lamina due to differing hygrothermal expansion of each lamina. The hygrothermal strains are not equal in a lamina in the longitudinal and transverse directions since the elastic constants and the thermal and moisture expansion coefficients of the fiber and matrix are different. In the following sections, stress-strain relationships are developed for unidirectional and angle lamina subjected to hygrothermal loads.

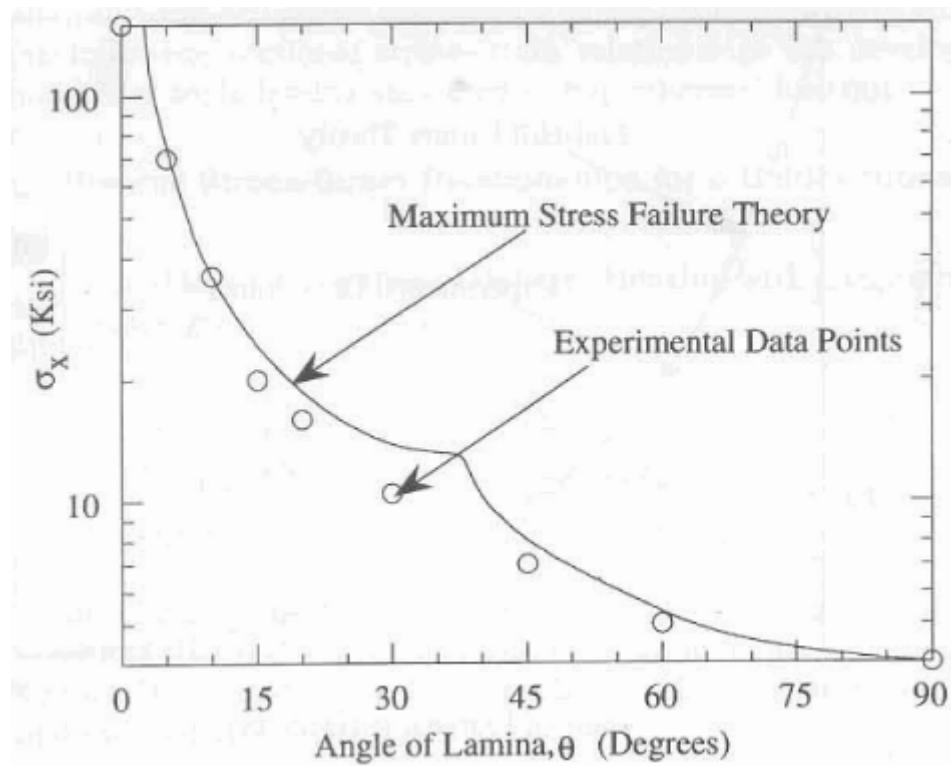


Figure M10.3.6 Maximum normal tensile stresses in the x-direction as a function of angle of lamina using Maximum Stress failure theory.

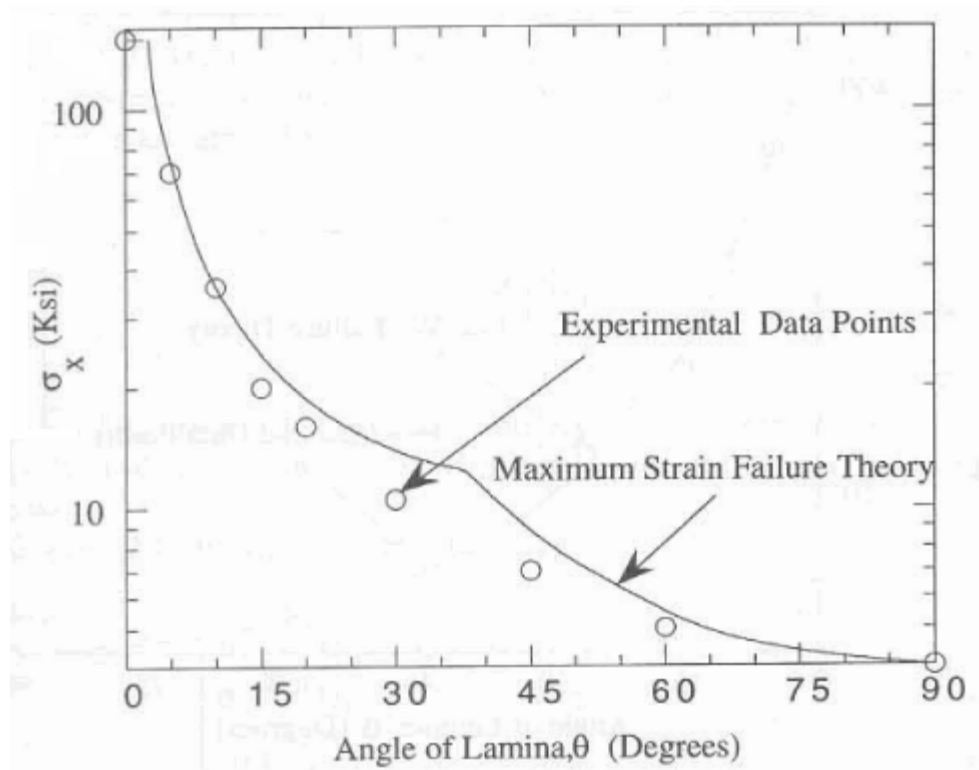


Figure M10.3.7 Maximum normal tensile stress in the x-direction as a function of angle of lamina using Maximum Strain failure theory.

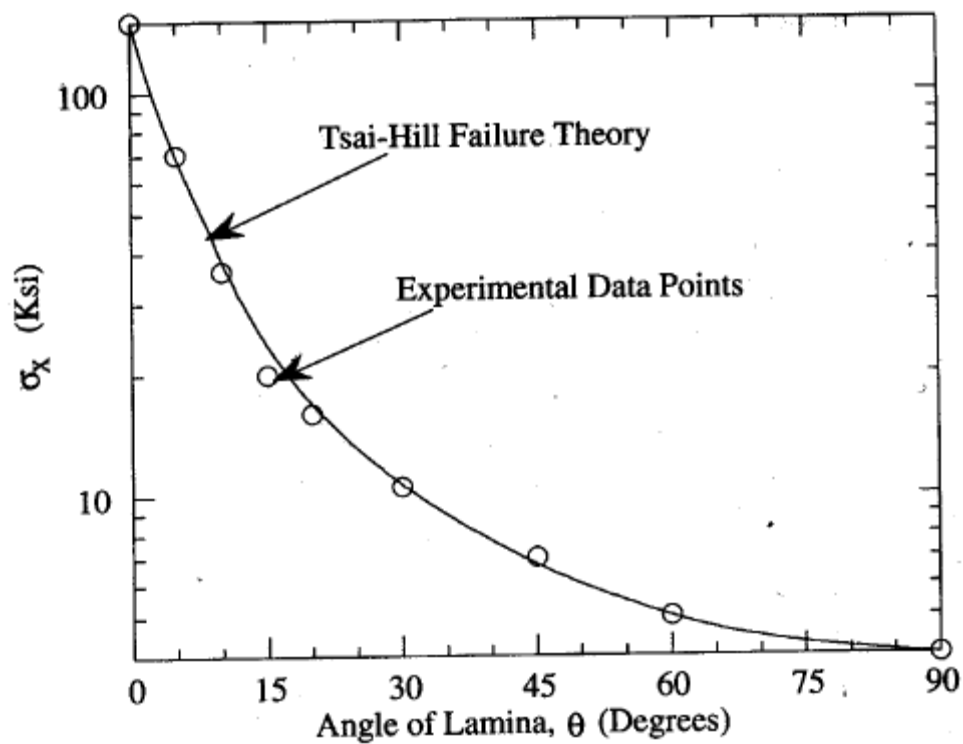


Figure M10.3.8 Maximum normal tensile stresses in the x-direction as a function of angle of lamina using Tsai-Hill failure theory.

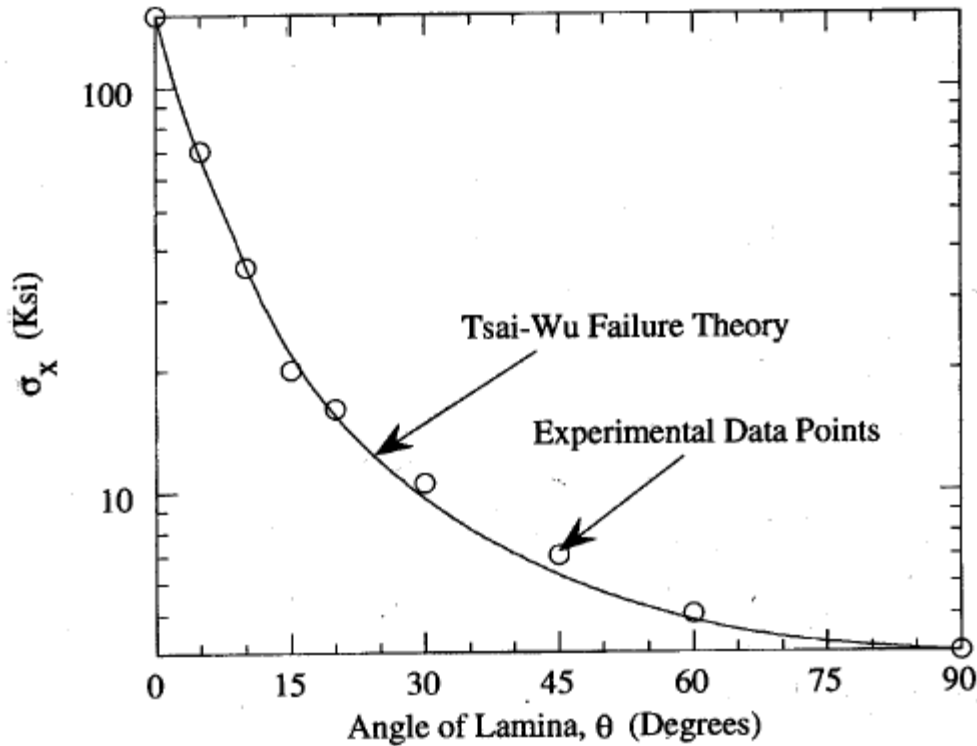


Figure M10.3.9 Maximum normal tensile stresses in the x-direction as a function of angle of lamina using Tsai-Hill failure theory.

M10.3.11.1 Hygrothermal Stress-Strain Relationships for a Unidirectional Lamina

For a unidirectional lamina, the stress-strain relationship with temperature and moisture difference gives,

$$\begin{bmatrix} \epsilon_1 \\ \epsilon_2 \\ \gamma_{12} \end{bmatrix} = \begin{bmatrix} S_{11} & S_{12} & 0 \\ S_{12} & S_{22} & 0 \\ 0 & 0 & S_{66} \end{bmatrix} \begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} + \begin{bmatrix} \epsilon_1^T \\ \epsilon_2^T \\ 0 \end{bmatrix} + \begin{bmatrix} \epsilon_1^C \\ \epsilon_2^C \\ 0 \end{bmatrix} \quad (\text{M10.3.1})$$

where the subscripts T and C are used to denote temperature and swelling, respectively. Note that the temperature and moisture changes do not have any shearing strain terms, since no shearing strains are induced in the material axes. The thermal-induced strains are given by,

$$\begin{bmatrix} \epsilon_1^T \\ \epsilon_2^T \\ 0 \end{bmatrix} = \Delta T \begin{bmatrix} \alpha_1 \\ \alpha_2 \\ 0 \end{bmatrix} \quad (\text{M10.3.2})$$

where α_1 and α_2 are the longitudinal and transverse coefficients of thermal expansion, respectively, and ΔT is the temperature change. The moisture-induced strains are given by,

$$\begin{bmatrix} \epsilon_1^C \\ \epsilon_2^C \\ 0 \end{bmatrix} = \Delta C \begin{bmatrix} \beta_1 \\ \beta_2 \\ 0 \end{bmatrix} \quad (\text{M10.3.3})$$

where β_1 and β_2 are the longitudinal and transverse coefficients of swelling, respectively, and ΔC is the weight of moisture absorption per unit weight of the lamina.

Equation (M10.3.1) can be inverted to give

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \begin{bmatrix} \epsilon_1 - \epsilon_1^T - \epsilon_1^C \\ \epsilon_2 - \epsilon_2^T - \epsilon_2^C \\ \gamma_{12} \end{bmatrix} \quad (\text{M10.3.4})$$

M10.3.11.2 Hygrothermal Stress-Strain Relationships for an Angle Lamina

The stress-strain relationship for an angle lamina takes the following form:

$$\begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} = \begin{bmatrix} \bar{S}_{11} & \bar{S}_{12} & \bar{S}_{16} \\ \bar{S}_{12} & \bar{S}_{22} & \bar{S}_{26} \\ \bar{S}_{16} & \bar{S}_{26} & \bar{S}_{66} \end{bmatrix} \begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} + \begin{bmatrix} \epsilon_x^T \\ \epsilon_y^T \\ \gamma_{xy}^T \end{bmatrix} + \begin{bmatrix} \epsilon_x^C \\ \epsilon_y^C \\ \gamma_{xy}^C \end{bmatrix} \quad (\text{M10.3.5})$$

Where

$$\begin{bmatrix} \epsilon_x^T \\ \epsilon_y^T \\ \gamma_{xy}^T \end{bmatrix} = \Delta T \begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_{xy} \end{bmatrix} \quad (\text{M10.3.6})$$

and .

$$\begin{bmatrix} \epsilon_x^C \\ \epsilon_y^C \\ \gamma_{xy}^C \end{bmatrix} = \Delta C \begin{bmatrix} \beta_x \\ \beta_y \\ \beta_{xy} \end{bmatrix} \quad (\text{M10.3.7})$$

The terms α_x , α_y , and α_{xy} are the coefficients of thermal expansion for an angle lamina and are given in terms of the coefficients of thermal expansion for a unidirectional lamina as

$$\begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_{xy}/2 \end{bmatrix} = [T]^{-1} \begin{bmatrix} \alpha_1 \\ \alpha_2 \\ 0 \end{bmatrix} \quad (\text{M10.3.8})$$

Similarly, β_x , β_y and β_{xy} are the coefficients of moisture expansion for an angle lamina and are given in terms of the coefficients of moisture expansion for a unidirectional lamina as

$$\begin{bmatrix} \beta_x \\ \beta_y \\ \beta_{xy}/2 \end{bmatrix} = [T]^{-1} \begin{bmatrix} \beta_1 \\ \beta_2 \\ 0 \end{bmatrix} \quad (\text{M10.3.9})$$

From Equation (M10.3.1), if there are no constraints placed on a lamina, no mechanical strains will be induced in it. This also implies, then, no mechanical stresses are induced. But in a laminate, even if there are no constraints on the laminate, the difference in the thermal/moisture expansion coefficients of the various layers induces different thermal/moisture expansions in each layer. This difference results in residual stresses and will be explained fully in section M10.3.11.4.

M10.3.11.4 HYGROTHERMAL EFFECTS IN A LAMINATE

In Section M10.3.11.3, the hygrothermal strains were calculated for an angle and uni-directional lamina subjected to a temperature change, ΔT , and moisture content change, ΔC . As mentioned, if the lamina is free to expand, no residual mechanical stresses would develop in the lamina at the macro-mechanical level. However, in a laminate with various plies of different angle or material, each individual lamina is not free to deform. This results in residual stresses in the laminate.

M10.3.11.4.1 Hygrothermal Stresses and Strains

Sources of hygrothermal loads include cooling down from processing temperatures, operating temperatures different from processing temperatures, and humid environment such as an aircraft flying at high altitudes. Each ply in a laminate gets stressed by the deformation differences of adjacent lamina. Only the strains which are in excess of or less than the hygrothermal strains in the unrestricted lamina produce the residual stresses. These strain differences are called mechanical strains and the stresses caused by them are called mechanical stresses.

The mechanical strains induced by hygrothermal loads alone,

$$\begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix} = \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix} - \begin{bmatrix} \epsilon_x^T \\ \epsilon_y^T \\ \gamma_{xy}^T \end{bmatrix} - \begin{bmatrix} \epsilon_x^C \\ \epsilon_y^C \\ \gamma_{xy}^C \end{bmatrix} \quad (\text{M10.3.10})$$

where the superscript 'M' represents the mechanical strains, 'T' stands for the free expansion thermal strain, and 'C' refers to the free expansion moisture strains. Using stress-strain Equation $E_2 = \sigma_2 / \epsilon_2 = 1 / S_{22}$, the hygrothermal stresses in a lamina are then given by

$$\begin{bmatrix} \sigma_x^{TC} \\ \sigma_y^{TC} \\ \gamma_{xy}^{TC} \end{bmatrix} = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix} \quad (\text{M10.3.11})$$

where 'TC' stands for combined thermal and moisture effects. Hygrothermal stresses induce zero resultant forces and moments in the laminate in the laminate and hence in the n-ply laminate shown in Figure M10.3.10,

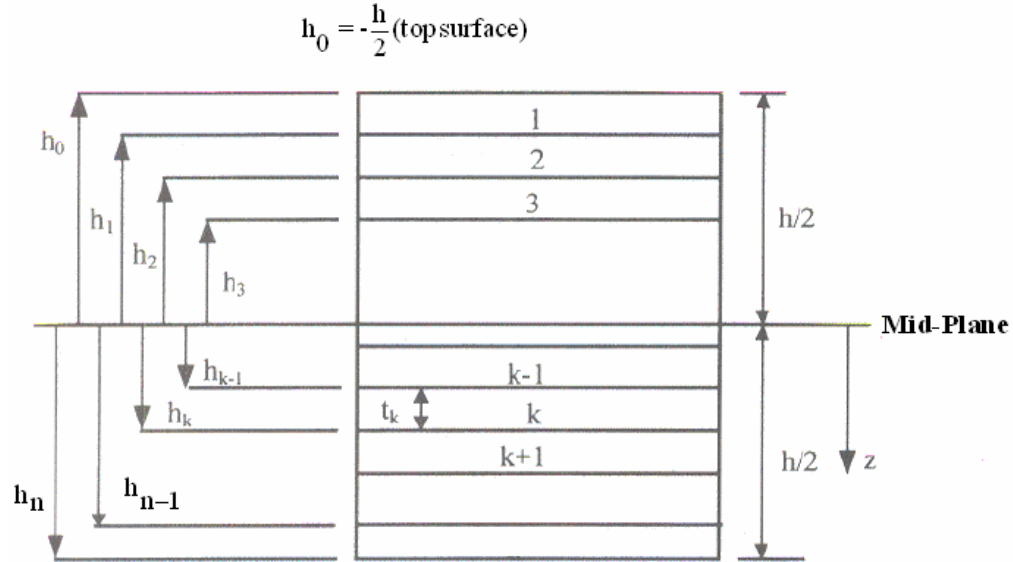


Figure M10.3.10 Coordinate locations of plies in a laminate

$$\int_{-h/2}^{h/2} \begin{bmatrix} \sigma_x^{TC} \\ \sigma_y^{TC} \\ \tau_{xy}^{TC} \end{bmatrix} dz = 0 = \sum_{k=1}^n \int_{h_{k-1}}^{h_k} \begin{bmatrix} \sigma_x^{TC} \\ \sigma_y^{TC} \\ \tau_{xy}^{TC} \end{bmatrix}_k dz \quad (\text{M10.3.12})$$

$$\int_{-h/2}^{h/2} \begin{bmatrix} \sigma_x^{TC} \\ \sigma_y^{TC} \\ \tau_{xy}^{TC} \end{bmatrix} z dz = 0 = \sum_{k=1}^n \int_{h_{k-1}}^{h_k} \begin{bmatrix} \sigma_x^{TC} \\ \sigma_y^{TC} \\ \tau_{xy}^{TC} \end{bmatrix}_k dz \quad (\text{M10.3.13})$$

From Equations (M10.3.11) to (M10.3.13),

$$\sum_{k=1}^n \int_{h_{k-1}}^{h_k} \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix} dz = 0 \quad (\text{M10.3.14a})$$

and

$$\sum_{k=1}^n \int_{h_{k-1}}^{h_k} \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix} z \, dz = 0 \quad (\text{M10.3.14b})$$

On substituting Equations (M10.3.10) and (M10.3.15), they give

$$\begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^0 \\ \epsilon_y^0 \\ \gamma_{xy}^0 \end{bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix} = \begin{bmatrix} N_x^T \\ N_y^T \\ N_{xy}^T \end{bmatrix} + \begin{bmatrix} N_x^C \\ N_y^C \\ N_{xy}^C \end{bmatrix} \quad (\text{M10.3.15})$$

$$\begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^0 \\ \epsilon_y^0 \\ \gamma_{xy}^0 \end{bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix} = \begin{bmatrix} M_x^T \\ M_y^T \\ M_{xy}^T \end{bmatrix} + \begin{bmatrix} M_x^C \\ M_y^C \\ M_{xy}^C \end{bmatrix} \quad (\text{M10.3.16})$$

The four arrays on the right-hand side of the above Equations (M10.3.16) and (M10.3.17) are given by,

$$[N^T] = \begin{bmatrix} N_x^T \\ N_y^T \\ N_{xy}^T \end{bmatrix} = \Delta T \sum_{k=1}^n \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_{xy} \end{bmatrix}_k (h_k - h_{k-1}) \quad (\text{M10.3.17})$$

$$[M^T] = \begin{bmatrix} M_x^T \\ M_y^T \\ M_{xy}^T \end{bmatrix} = \frac{1}{2} \Delta T \sum_{k=1}^n \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{bmatrix} \alpha_x \\ \alpha_y \\ \alpha_{xy} \end{bmatrix}_k (h_k^2 - h_{k-1}^2) \quad (\text{M10.3.18})$$

$$[N^C] = \begin{bmatrix} N_x^C \\ N_y^C \\ N_{xy}^C \end{bmatrix} = \Delta C \sum_{k=1}^n \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{bmatrix} \beta_x \\ \beta_y \\ \beta_{xy} \end{bmatrix}_k (h_k - h_{k-1}) \quad (\text{M10.3.19})$$

$$[M^C] = \begin{bmatrix} M_x^C \\ M_y^C \\ M_{xy}^C \end{bmatrix} = \frac{1}{2} \Delta C \sum_{k=1}^n \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{bmatrix} \beta_x \\ \beta_y \\ \beta_{xy} \end{bmatrix}_k (h_k^2 - h_{k-1}^2) \quad (M10.3.20)$$

The loads in Equations (M10.3.18) to (M10.3.25) are called fictitious hygrothermal loads and are known. One can calculate the midplane strains and curvatures by combining Equations (M10.3.17) and (M10.3.18), which is

$$\begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \epsilon_x^0 \\ \epsilon_y^0 \\ \gamma_{xy}^0 \end{bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix} = \begin{bmatrix} M_x^T \\ M_y^T \\ M_{xy}^T \end{bmatrix} + \begin{bmatrix} M_x^C \\ M_y^C \\ M_{xy}^C \end{bmatrix} \quad (M10.3.21)$$

Using Equation,

$$\begin{Bmatrix} \epsilon_x \\ \epsilon_y \\ \epsilon_z \end{Bmatrix} = \begin{Bmatrix} \epsilon_x^0 \\ \epsilon_y^0 \\ \epsilon_z^0 \end{Bmatrix} + z \begin{Bmatrix} \kappa_x \\ \kappa_y \\ \kappa_z \end{Bmatrix}$$

One can calculate the global strains in any ply of the laminate. These global strains are the actual strains in the laminate. However, it is the difference between the actual strains and the free expansion strains, which results in mechanical stresses. The mechanical strains in the k^{th} ply are given by Equation (10.3.10) as

$$\begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix}_k = \begin{bmatrix} \epsilon_x \\ \epsilon_y \\ \gamma_{xy} \end{bmatrix}_k - \begin{bmatrix} \epsilon_x^T \\ \epsilon_y^T \\ \gamma_{xy}^T \end{bmatrix}_k - \begin{bmatrix} \epsilon_x^C \\ \epsilon_y^C \\ \gamma_{xy}^C \end{bmatrix}_k \quad (M10.3.22)$$

The mechanical stresses in the k^{th} ply are then calculated by

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix}_k = \begin{bmatrix} \bar{Q}_{11} & \bar{Q}_{12} & \bar{Q}_{16} \\ \bar{Q}_{12} & \bar{Q}_{22} & \bar{Q}_{26} \\ \bar{Q}_{16} & \bar{Q}_{26} & \bar{Q}_{66} \end{bmatrix}_k \begin{bmatrix} \epsilon_x^M \\ \epsilon_y^M \\ \gamma_{xy}^M \end{bmatrix}_k \quad (M10.3.23)$$

The fictitious hygrothermal loads represent the loads in Equations (M10.3.17) to (M10.3.20) which one can apply mechanically to induce the same *stresses* and strains as by the hygrothermal load. Hence if both mechanical and hygrothermal loads are applied, one can add the mechanical loads to the fictitious hygrothermal loads to find the ply-by-ply stresses and strains in the

laminate, or one can separately apply the mechanical and hygrothermal loads and then add the resulting stresses and strains from the solution of the two problems.

Learning Unit-4: M10.4

M10.4 Recycling Of Composite Materials

Recycling is one of the biggest issues facing the composites industry, particularly for large-volume applications. Increasingly stringent environmental regulations are likely to restrict the use of composites in favour of materials that can be recycled cost effectively. This section deals with composite matrices that cannot be commercially depolymerised into monomer.

M10.4.1 Categories of Scrap Composites

Scrap composites can be conveniently divided into three categories:

1. Scrap in the form of **offcuts**, **rejects**, **sprues** etc. arising in the manufacture of composite products. Increasingly, this waste material is used in primary recycling by blending it as filler or reinforcement with **virgin plastic** of the same chemical origin. This route is subject to the careful control of the levels of contamination in the comminuted composite and to the deterioration in physical properties which may be caused by repeated thermal and mechanical processing. Some thermosets have been successfully recycled since the mid 1980's without adverse effect on quality.
2. Single grades of contaminated plastic collected from consumers or processors may provide feedstock for primary or secondary recycling, subject to the feasibility of contaminant removal.
3. Mixtures of two or more grades of composite compounds arise as industrial or consumer scraps. This category of scrap poses a substantial problem in composites recycling, because of the problems associated with automatic identification, separation and determination and control of composition. Tertiary recycling is expected to be the more appropriate route for recycling.

M10.4.2 Recycling Methods for: Thermoplastic, Thermoset and Metal Matrix Composites

M10.4.2.1 Thermoplastic Matrix Composites

Thermoplastic composites scrap arising in the first two categories provides for more straightforward recycling than thermoset composites, principally because the thermoplastic can be melted. **Fibre attrition** and **degradation** of the matrix polymers lead to reuse applications with less-demanding physical property requirements. A good example of large volume thermoplastic composite recycling is long-glass-fibre-mat reinforced polypropylene (GMT). The offcuts from GMT sheet used for thermoforming of products particularly in the manufacture of automotive parts can be subsequently used after comminution and used over again as raw material for semi-finished sheet. Additionally, the offcuts or reject parts can be ground and used for extrusion or for injection moulding.

M10.4.2.2 Thermoset Matrix Composites

The recycling of thermoset composites presents great difficulties, centered around the irreversibility of cross-linking, the fibre attrition associated with comminution, and a polymer content that may be less than 30% of the total weight. The bulk of the material is often glass-fibre reinforcement or filler, including fire retardants and resin dilutants. In tertiary recycling, the problem is not just one of recycling the polymer.

Greater stability in the supply of scrap is associated with large sources of standardized composite scrap arising from cooperative industry ventures. For example, ERCOM, consortium of four large European composite manufacturers in partnership with a number of leading raw materials suppliers, shred components manufactured from polyester and vinyl-ester-based sheet and bulk moulding compounds to a range of well-defined particle sizes. The resultant fibre and powder fractions can be used in the production of new **bulk-moulding compound (SMC/BMC)** components, and can also be used as reinforcing material for thermoplastics and other materials.

This type of initiative is providing for large-scale utilization of composites scrap and the incentive for the development of more comprehensive secondary and tertiary recycling operations. Considerable imagination is evident in the research and development work which is in progress to identify profitable routes for the recycling of composite materials. These include recycling machinery development, with attention to comminutive procedures, and new product development that has to overcome the technical, cost and aesthetic advantages of competing virgin materials. Recent research relates to proposed radical and potentially very substantial disposal options for composites scrap. Combustion with heat recovery is proposed as a route for utilizing the energy content of the matrix polymer. The behaviour of a range of composites during combustion, the form of the ash product and the emissions during combustion has been investigated systematically.

Two industrial processes that utilize the energy content and the inert materials arising in the ash have been proposed. In cement manufacture, **thermoset composites** may be burned in a cement kiln to utilize their energy content and the mineral materials utilized in the **cement klinker**. Alternatively, polymeric materials filled with calcium carbonate may be of use as a fuel substitute and sulphur oxide removing agents in coal-fired fluidized bed combustion. Large volume applications for selected recycled composites are being developed as shot-blasting media for selective paint removal, and for use as soil conditioners.

Thermoset composite materials represent a large percentage of composites manufacturing, particularly in the automotive sector, and bulk-moulding compound (BMC/SMC's) have been the subject of considerable research and success in recycling.

M10.4.2.3 Metal Matrix Composites

The term **metal matrix composite (MMC)** encompasses a wide range of materials. Common to all these is a continuous metallic matrix material and the reinforcing phase is usually a ceramic. MMC's are categorized according to the **morphology** of the reinforcement i.e. **continuous fibers, whiskers, or particles**, as shown in Figure M10.4.2.3.

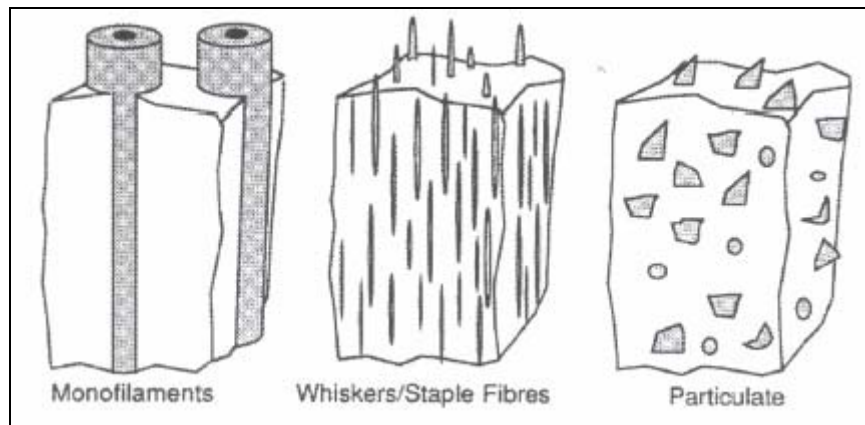


Figure M10.4.2.3 Morphology of the Reinforcement

The high **formability** of the metallic matrix gives MMC's an advantage over polymeric materials, and they are often available in **stock form** (e.g. billets, rods and tubes), from which components are shaped and formed during secondary operations. Another advantage is that despite being **anisotropic**, excellent axial performance can be combined with transverse properties which are more than satisfactory.

M10.4.2.3.1 History

Modern MMC's first appeared in the 1920's with the production of aluminium/alumina dispersion hardened systems. The 1950's saw the development of precipitation hardened materials and in both these materials, the presence of small particles impedes dislocation movement and enhances the toughness of the base metal. Also, small percentages of filler are required (<15%) to obtain this enhancement. Creep is effectively suppressed in these materials as dislocations must climb over the dispersoids by **diffusive processes** and this results in **creep** rates decreasing with increasing dispersoids size.

Evolution continued through the 1970's with the introduction of **dual phase** steels, which are effectively particulate MMC's consisting of up to 20% **martensite** within a soft **ferrite** matrix. This form of steel is regarded as the forerunner of modern MMC technology. Fibrous MMC's were developed in the 1960's and were based on tungsten or boron fibres embedded within a copper or aluminium matrix. The morphology of these materials is similar to that of polymer matrix composites however, interest declined as production costs escalated. Nowadays, interest in titanium matrix composite has provided something of a renaissance in this form of material.

The most modern developments in MMC technology have been in the use of whisker reinforcements. The combination of good transverse properties, low cost, high workability and significant increases in performance over unreinforced alloys has made them currently the most commercially attractive system for many different applications.

M10.4.2.3.2 Fabrication Processes

In modern MMC technology, there are four main routes taken in the manufacture of an MMC component:

1. **Primary liquid phase processing:** This classification encompasses techniques such as squeeze casting and squeeze infiltration; spray deposition; slurry casting and reactive processing.
2. **Primary solid state processing:** This includes powder blending and pressing; diffusion bonding of foils and physical vapour deposition.
3. **Secondary processing:** e.g. extrusion and drawing; rolling; forging; isostatic pressing; superplastic processing and sheet forming.
4. **Machining and forming processes:** such as electrical, mechanical and fluid-jet cutting, and joining. These processes are rather complex and outside the scope of this course.

M10.4.2.3.3 Applications

Perhaps one of the earliest MMC applications was in the Space Shuttle, in which a structural component in the cargo bay section framework was made from a 60% boron **monofilament reinforced** aluminium-based composite. Other major applications are based primarily on the enhanced stiffness and creep characteristics of MMC materials e.g. engine components, drive shafts, bicycle frames and cross-booms on yachts.

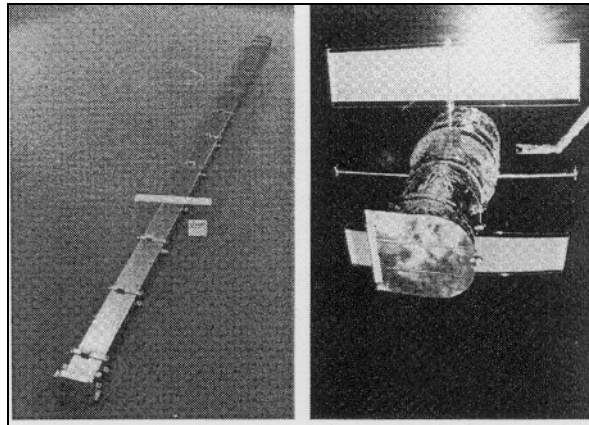


Figure M10.4.3.2.3.3 (a) Photograph of a satellite boom/waveguide structure fabricated from an aluminium-carbon fibre MMC. The fibres are aligned parallel to the axis of the boom.



Figure M10.4.3.2.3.3 (b) Photograph of a rotor brake disc, made from Duralcan[®] cast Al-10wt%Si / 20vol% SiC.



Figure M10.4.3.2.3.3 (b) Photograph of the Stump-jumper M2 bicycle, the frame is made from Duralcan[®] 6061 Al-10wt% alumina