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REVIEWS

Design Allowable Considerations for use of Laminated Composites in Aircraft Structures

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Abstract | Carbon fibre polymer composites have evolved over the years to become major structural materials for primary structures of the aircraft today. The paper reviews the work carried out over last four decades on carbon fiber polymer composites to create an understanding of their behaviour in order to set up a philosophy of design and arriving at design allowable values for strength so as to ensure safety and performance of aircraft with minimum weight penalty. The rationale behind the choice of allowable values and the process to arrive at them is explained. While several issues are discussed, three major aspects are emphasised: the environmental (hygrothermal) effect, the effect of holes and fasteners and impact damage. The directions of the current improvements and course of future developments are indicated in relation to their influence on the design allowables.

Keywords: *Carbon Fibre Composites, Laminates, Aircraft Structures, Design Allowable, Structural Design, Impact Damage, Holes in composites, Fastener joints, Hygrothermal effect.*

1 Introduction

Over the past few decades Fibre Reinforced Polymer Matrix Composites have emerged as strong contenders for building load bearing structures giving a tough competition to the conventional structural materials such as aluminium alloys and steels in several engineering sectors such as aerospace, construction, transportation, off-shore structures and others. In particular, in the aeronautical sector where light-weighting is a major issue, the composite usage in aircraft structure has graduated from just being marginal and that too in tertiary and secondary structures in 1970's to being the preferred material for large primary structures in modern frontline advanced aircraft in both combat and transport category. To quote a few examples: Airbus 320 , Boeing 737, then Boeing 777, 787 and Airbus 380, FA-18, Grippen, Eurofighter, French Rafale and Indian Tejas Light Combat Aircraft—all have seen a large scale use of composites in primary structures. See, for example, Figure 1 showing use of carbon/epoxy composites in LCA.

The major form of composites that pioneered the large scale usage in aircraft structures has been the laminated carbon epoxy composites. Several aspects of the behaviour of these materials have been extensively investigated over the years. On the theoretical or analysis side, the investigations have seen a great evolution of methods and techniques starting with the orthotropic elasticity,[1–4] the development of the laminate theory,[5–9] the thin and thick plate and shell theories,[8] the use of finite element modelling and analysis, and the use of fracture mechanistic concepts and damage progression models.[10,11] Similarly, experimental and test techniques have also seen a great evolution[12] and helped understand the behaviour of composites under various conditions, provide validation of theoretical models and generate data which can be used for designing the composite structures. Much of this knowledge has been documented in excellent reports, books and handbooks[13–17] and a significant part has already become text-book material today, see for example.[18–20]

The complexity of the composites behaviour and the fact of it being so very different from the conventional materials such as aluminium alloys led to several investigations and studies in the initial period for setting up a "philosophy" of

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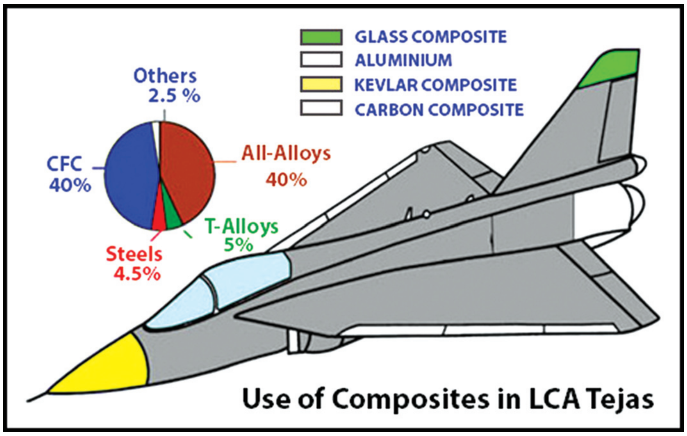

Figure 1: Use of composites in LCA airframe.
(Source: http://www.tejas.gov.in/images/content/technology/composite_materials.jpg)

design for load carrying aircraft structures using the laminated composites. One of the interesting debates in this context has been around the choice of values of strength and stiffness (and a few other physical parameters) to be used as allowable values for the design in order to ensure safety of operation of the aircraft with desired performance and durability. The concurrent issues of how to determine or derive such "design allowable" values, i.e., test techniques and analysis methods of arriving at such values, and ultimately of how to verify and validate (or certify) the structures so designed and built have also seen extensive examination. An overview of the considerations for arriving at the design allowables forms the topic of this paper.

It is perhaps obvious that the design allowable values would be influenced by the features of the material behaviour as well as the features of the intended structure and its intended performance. In the present paper, we start with indication of various factors and issues that need to be considered and then look into some important ones amongst them. Amongst the various types and forms of composites, the laminated composites using carbon fibres of modulus around 230 GPa and strength around 3.2 GPa (often referred to as "Standard" Modulus Carbon fibre, such as T300 of Toray) with either 120°C or 170°C curing epoxy resin systems has been the main work-horse material for composite studies and usage over several decades and this forms the basis for our discussions in this paper. We will also briefly examine later advances such as improved materials (e.g., Intermediate Modulus Carbon fibre and the use of toughened resins as matrices) and modelling techniques in the context of their influence on the way we may choose design allowables. We conclude with a few remarks on some outstanding issues and line of future work.

2 Nature of Composite Behaviour

2.1 *Material behaviour features*

The choice of laminated carbon-fibre-reinforced-polymer composites (CFRP) as against the conventional aluminium alloys for airframe structures is dictated by the need for light-weighting the structure. While this is the prime reason for the choice, this is not the only reason, nor is this the compelling one. Amongst others are

a) *the need for tailorability* of wing-like aerodynamic structures which need tailored flexibility over a large area and which can be achieved with comparative ease using the laminated composites
b) the ability of the composites for *large part integration* and
c) *the excellent fatigue resistance* of composites for in-plane loading, the type of loading that the aircraft semi-monocoque, thin-walled structures are often required to carry.

It is important to keep track of these objectives while selecting the design allowables, as more often than not a designer is required to trade off some of these advantages against some of the concerns about composites. Important amongst such concerns are:

a) *Hygrothermal Degradation*: The polymer matrix (epoxy) absorbs moisture and is prone to be affected by hygrothermal environment. Thus, there is a strong influence of environment on resin dominated material properties and behaviour.

b) *Stress Concentration around Holes and cut-outs*: Composites are prone to high stress concentration and tend to have a brittle failure. Since, often, the holes (such as for fasteners) and cut-outs (such as for access, or conduits) cannot be avoided, due care needs to be exercised in accounting for the stresses around holes and cut-outs.
c) *Delamination and Impact Damage*: The laminated structure of composite has "weak" interfaces which are prone to delaminate under relatively small peel stresses. This also makes composites very prone to impact damage. That such damage is often not seen on the surface and remains hidden is a major cause for concern.
d) *Variability of Properties*: Unlike the metallic structures, the composite material achieves its final material form only when the structural component itself is made. Thus, the material properties in the component are process dependent. A tight control is required on processing and tooling parameters in order to limit the variability in the properties to an acceptable level.
e) *Electrical Conductivity*: The carbon fibres conduct electricity, but with high resistance. This has implications on lightning protection measures, EMI/EMC effects and galvanic corrosion of interfacing metallic (especially aluminium) parts.

The last mentioned issue about the electrical conductivity is normally dealt through novel means of design features[21,22] and do not directly impinge upon the choice of allowable mechanical properties and will no more be discussed in this paper. The "variability" is an important consideration in achieving the desired probability of survival and becomes a part of the statistical calculations. The confidence levels for achieving the reliability and safety are usually prescribed by the regulatory authorities and the statistical procedures to account for the variability and scatter are well known and documented, see for example, MIL HDBK 17F[13,15] and MIL HDBK 5.[23] We will not discuss this further except to state that the industry practice in tune with the requirements posed by the regulatory authorities is to use A-basis material allowables (i.e., 99% probability of survival with 95% confidence) for structures or zones where there are no alternate load paths and the failure can be catastrophic and to use B-basis (i.e., 90% probability with 95% confidence) where there are alternate load paths and the failure is not catastrophic. Even as the procedures for calculating these basis values are well known[13,23–25] for various types of data distributions (such as normal, Weibull or log-normal), one needs to exercise care in ascertaining the applicable data distribution, the number of samples and factors which may make the tests invalid.

The focus in this paper is on discussing the rationale and considerations in handling the other three factors, namely, environmental effect of hygrothermal degradation, holes and fasteners, and impact damage and delaminations, in conjunction with various aspects of structural behaviour to arrive at design allowables.

2.2 *Structural behaviour features*

In addition to the features of the material behaviour mentioned above, the choice of allowables may often be influenced by the constructional features of the structural design such as ply-drops (or ply-terminations) required for thickness changes and for stiffness tailoring of aerodynamic structures such as wings, and fastener joints and T-joints (often without fasteners) required for connections such as spar-to-skin or stiffener-to-skin or to bulkheads. It is important to note that thin-walled construction of aircraft structures also needs consideration of structural stability (buckling) in addition to the strength and stiffness considerations. An important aspect of the behaviour of CFRP is that they have multiple possible failure modes depending on the initial material quality, the structural configuration and the load flows, such as, fibre-matrix interface debonds, fibre pull-out in tension, fibre buckling in compression, matrix cracking; matrix failure in shear or in tension, delamination, fibre-breaks, etc. Many of them such as matrix cracking or interface debonds, often remain at sub-critical level and do not lead to substantial loss in strength or stiffness or the failure of structure. Thus, it is important to arrive at a proper definition or criterion of "failure" depending upon the performance and safety requirements as this has a significant influence on the "allowable" values. As the fibres are the primary load carrying constituents in the composite, fibre failure modes are the primary failure modes (critical failure modes) influencing the strength and stiffness; the other modes may not directly influence the immediate load carrying capability (therefore "sub-critical") but may lead to degradation of structural performance over a period of time and need to be watched. The ability of the fibres to sustain loads, particularly in compression and shear, is greatly influenced by the support they receive from the matrix and any loss of support due to matrix or interface failure needs to be accounted for. In metallic materials, cracks open up under tensile stress and close under compressive stress and thus in general tensile stresses are considered to be more critical than the compressive ones. However, in laminated composites, delaminations tend to open up in compression. Moreover, the micromechanics of composites shows that any loss of support for the fibres even at micro level will result in premature fibre-micro-buckling, thus aiding overall compressive failure. On the other hand, the fibres in composites can carry large loads in tension till the fibre failure. Thus, often, compressive loading becomes more critical for the composites. The internal damages such as delaminations and impact damage tend to progress in compression and this coupled with micro-buckling of fibres and the overall buckling of the structure adds to the complexity of handling compressive loading. The failure in a laminated composite is often progressive, starting with the failure of one or more plies and progressively spreading to other plies. However, the normal industry practice presently is to use the first-ply failure as the basis for design. Utilising the strength beyond the first-ply failure continues to be a topic for further investigations on composites.

2.3 Laminate behaviour: Laminate plate theory and failure theories

The laminated plate theory is very well developed and understood today and will not form part of discussion in this paper. Similarly, a number of failure theories have been evolved over the years and a good review of them can be found in several books and monographs.[5–9,26] Suffices it to remark here that the basic unidirectional lamina strength and stiffness properties form the basis for deriving the properties of a laminated composite. The material allowables are derived for basic unidirectional lamina properties and failure strength envelopes in terms of carpet plots, for multiangular laminates are created by using laminate theory and an appropriate failure theory, (see for example,[27]), which may be validated by tests on typical multiangular laminates of interest. These aspects are well established and can be found in various texts on composites referred earlier. On the other hand, while analysing a structure, one works out ply level strains and stresses from the laminate stresses or strains and applies the failure theory for assessing the margin of safety based on lamina strength.

2.4 Laminate lay-up

While many studies were conducted on Unidirectional (UD), Cross-Ply (CP) and Quasi-Isotropic (QI) laminates, they are not the most efficient construction and do not really use the directional tailorability advantage of composites. On the other hand, arbitrarily optimized lay-ups can lead to unsymmetric, unbalanced laminates which have more complex behaviour that is difficult to

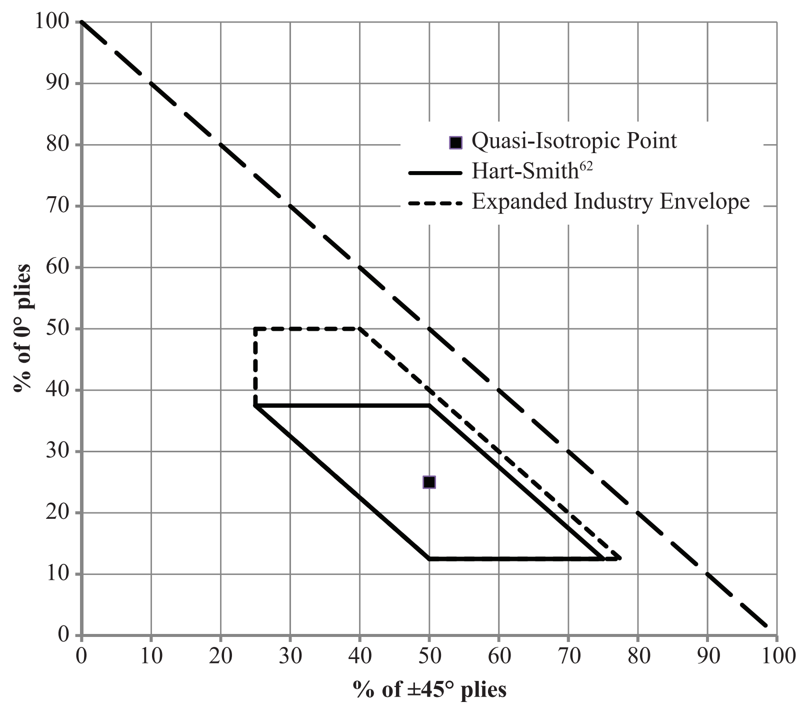

Figure 2: Envelope of practically useable laminates, 0, ±45, 90 family.

predict and therefore do not inspire designers' confidence. Thus, most of the useful combinations of laminate lay-ups are restricted to being symmetric, balanced with orientations of 0, ±45° and 90° with adequate plies in any one direction to ensure integrity. Based on stress concentration considerations, Hart-Smith[38,40] had proposed an envelope of laminate configurations (see Fig. 2) around the QI configuration ensuring at least 12.5% plies in any direction. With improved understanding and analytical capabilities, the industry has progressed to relaxing some of these requirements, such as using a somewhat larger envelope of laminate (see Fig. 2) configurations and using unbalanced (but not unsymmetric, due to concern about warping during manufacturing) laminates and using other orientations such as 30° and 60°. However, for the discussions in this paper, we will restrict to 0, ±45° and 90° orientations family and symmetric laminates.

3 Environmental Effects

3.1 *Hygrothermal degradation—hot-wet effect*

By far the most important effect of the environment that affects the design allowables is the hygrothermal degradation of the properties due the moisture absorption by the polymeric matrix, see Figure 3. The widely used epoxy based matrices can absorb up to 4–6% moisture by weight which translates to about 1.4–2.0% moisture absorption in carbon fibre composites. This degrades the matrix dominated behaviour, such as transverse strengths as well as shear and compression strengths. High temperatures close to the glass transition (T_g) of the matrix softens the matrix. The plasticization of the matrix (due to moisture as well as due to high temperature) reduces support to the fibres and thus causes an overall degradation of mechanical properties in shear and compression. Also, the moisture reduces T_g of the matrix significantly (by as much as 50°C) and thus puts a limit on the service temperature of the composite, which is usually kept to be about 20°C below T_g. Thus, the hot-wet (HTW) behaviour of composites has become an important issue in deciding design allowables. The moisture absorption is a diffusion process, follows largely the Fick's law and is mostly reversible. The absorption and desorption rates depend largely upon the temperature, and the saturation level depends largely on the relative humidity in the environment. The phenomenon is rather slow, and in thick laminates the time for moisture saturation can be very long. Two major issues that crop up when deriving design allowables are: (a) what is the realistic moisture gain in composites when aircraft is in service and which should be taken for deriving design allowables reflecting the actual degradation and (b) how to derive hot-wet properties by accelerated tests and whether the test factors obtained on small coupon tests can be applicable to large components. An excellent compilation of issues involved and the investigative results can be found in[28–36] and a good discussion in the Chap 12 of MIL-HDBK-17-3.[15] We discuss below briefly the rationale for accounting for these environmental effects.

While taking the saturation level moisture absorption for deriving design allowables can be safe, it can be quite unrealistic and will impose undue weight penalties. Various studies on worldwide exposure of composites[28,33,36] have shown that the moisture gain in realistic structures can be about 1%. A NASA and US Army study[37] has shown that the ground based coupons are good enough to reflect the environmental degradation conditions of the actual service component on the aircraft. Further, the industry appears to have arrived at a practice of taking the equilibrium level of moisture at 85% RH (rather than the saturation level) as representative of the realistic service. Thus, the test coupons can be exposed to this constant RH environment for ageing. The ageing temperature is usually taken as 70°C (for 170°C curing epoxy systems), low enough to avoid any effects of high temperature exposure, but high enough to reduce ageing time to acceptable levels. The hot-wet (HTW) condition then refers to the samples aged as above and tested at the high service temperature (say, 80°–100°C, for 170°C curing systems). The test factors obtained on such aged samples can then be used for setting up design allowable values. The adequacy of such factors for full scale structure is then usually established through a building block approach involving testing of structural features, test boxes and actual components. Investigations by various researchers have generally shown that for the hot-wet conditions, the UD compression and shear strengths degrades by about 30% while the T_g reduces from 170°C to about 125°–130°C.

3.2 *Other environmental effects*

Compared to the high temperature conditions, the cold temperatures (up to –55°C) do not have much effect on the material properties, except for enhancement of the brittle behaviour to some extent. However, the validity of this assertion needs to be established for certification for the material system being used. Thermal expansion and also swelling due to moisture absorption do

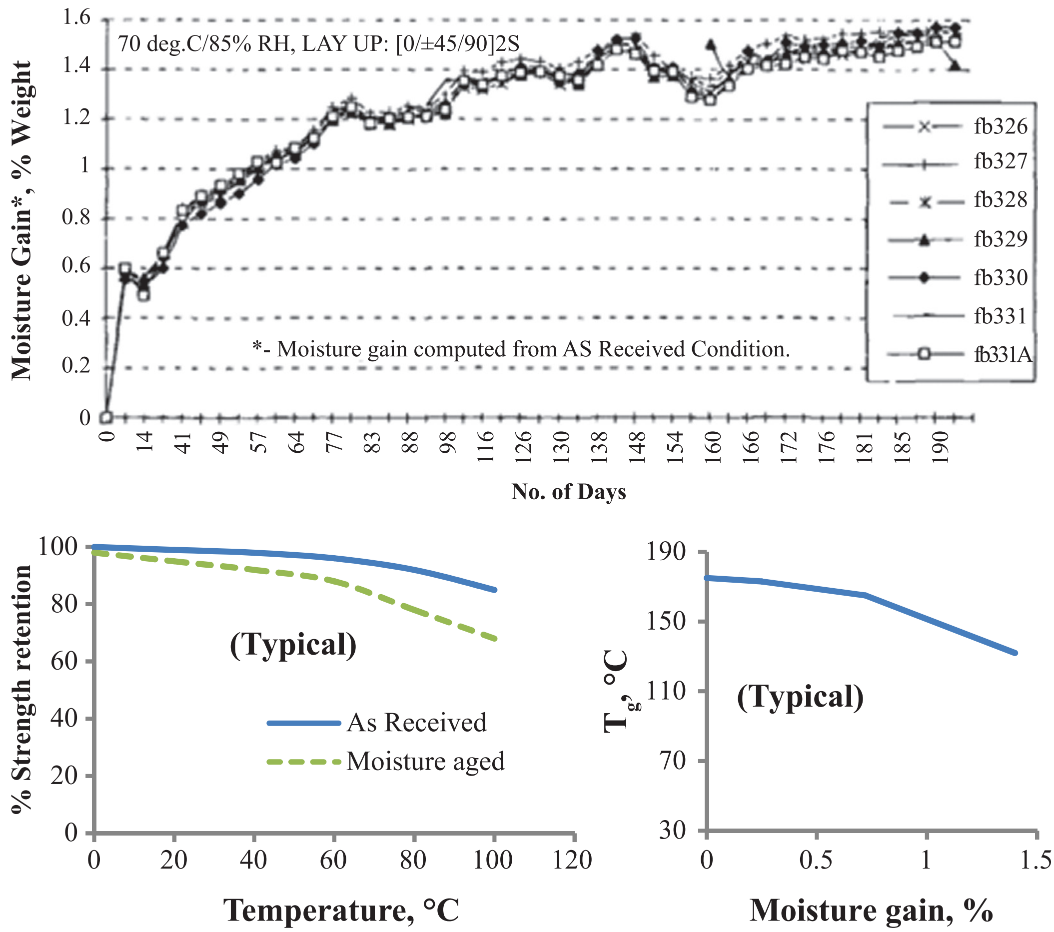

Figure 3: Hygrothermal effect in carbon epoxy composite. (a) Moisture ageing[71], (b) Strength degradation, (c) Lowering of T_g.

not form part of the design allowable exercise but need to be checked and accounted for in the design. There has been some concern and study about long term effect of radiation on the composite (especially UV radiation effect on epoxy resin) as well as effects of erosion due to sand and rain. These are generally catered for by use of suitable paints and protective coatings. One issue of significance to military fighter aircraft has been the effect of thermal spikes which some composite structures may be subject to due to weapon firing. Studies reported in[34,35] show that the direct effects of short duration spikes up to 140°C on compression or notched compression strengths are not very significant. There may be an indirect effect of several repeated spikes which may increase the moisture absorption to some extent.

4 Holes and Fastener Joints

4.1 *Historical perspective*

Due to the susceptibility of composites to stress concentration, holes and cut-outs need special care in designing with composites. Also, often, fastener joints cannot be avoided (and, in fact, may sometimes be preferred to bonded joints or integral parts) and one needs to work with knocked down allowable stresses in order to cater for such discontinuities. While in Al-alloys the stress concentration relief may be provided by local yielding, in the case of composites one needs to look for such relief by proper choice of lay-up. It is widely established that the addition of ±45° plies can provide such relief. Significant amount of studies were carried out in the decades of 1980's and 90's to understand the behaviour of composites with holes and fasteners, which formed the basis for arriving at design methodologies and design allowable values to be used. See for example, Refs.[37–76] A good account of these efforts can be found in AGARD report[74] with a historical perspective given by Oplinger.[75] Improvements over these early studies continue to interest researchers even today as seen from several studies being reported in literature; see, for example,[77–106] and also, a review of work on fastener joints.[95] Even the earliest studies clearly brought out that distinction must be made in treatment of open holes (or free, unloaded holes) and filled holes (such as in fastener joints), and also of those under tension as against those in compression. In addition to the stress concentration issues in the open holes case, which could be addressed largely by using anisotropic elasticity solutions,[2,4] the filled hole behaviour had to account for the contact around the hole interface between the plate and the pin, which made the problem nonlinear. An ingenious way of posing the problem in an inverse way (i.e., to find the load for a given contact configuration) proposed and used by Rao[52] and his co-workers[53,54] led to resolution of this issue. Using these concepts and later the Finite Element Method, Crews and Naik[61] studied the issue of bearing-bypass interaction which became an important aspect of joint design and deriving allowables. Studies and investigations on joints by Hart-Smith and co-workers,[38,40,46,47,60,62] as also by Collings and co-workers[39,44,49] and, Matthews and co-workers,[42,59,68,70,78] Gerharz and co-workers[55,69] and others over several years have contributed greatly to the understanding of various issues and how to handle them while designing with composites.

4.2 *Open holes: Tension*

One of the earliest studies on deriving practical design allowables is the one reported by Ekvall and Griffin[43] for the advanced Composites Fin and Aileron program which suggests knock-down factor for allowable for open hole tension to be 0.49 which is also supported by another study by Lafon.[57] Earlier, Hart-Smith[40] had argued that due to partial relief provided by matrix softening, the strength reduction may be limited to about 25% in quasi-isotropic laminates. Similar observation was made by Ruiz.[67] Combining experimental results and the analysis based on laminate theory, Bauer and Mennle[65] generated carpet plots for open hole tension, which suggested factors greater than 0.5. Experimental results of Schutz and Gerharz,[55] on various lay-ups have also shown factors in the range of 0.44 to 0.55. In tune with the above reported behaviour of composites and other similar results reported in several studies, the Industry has used open-hole tension factors of 0.4 to 0.5. Since such failure modes are fibre dominated, there is no further reduction necessary for hot-wet conditions. Because of the significant strength reduction, the open-hole tension case forms one of the driving factors for composite design.

4.3 *Open hole: Compression*

The strength reduction in open hole compression is not as severe as in tension, possibly because the neat compression strength itself is significantly less than in the tension and already accounts for some of the strength reducing features in compressive state of stress. Ekvall and Griffin[43] reported reduction by about 30%. Further, it is important to note that unlike the tension behaviour, compression behaviour is significantly affected by hot-wet conditions. Accounting for the hot-wet conditions, Potter and Purslow[45,50] noted that the failure is primarily governed by the instability of 0° plies and thus the lay-up sequence does not matter much. However, in presence of a lateral constraint (such as clamping force in a fastener joint), the fibres would buckle in-plane (rather than out-of-plane) and thus hot-wet effect may be reduced.

4.4 *Filled loaded holes, bearing strength*

The filled loaded holes are the most relevant features for fastener joints and the bearing strength becomes a very important parameter for design. The design philosophy for such joints revolves round avoiding lay-ups which may be excessively weak in any direction (in particular, in shear out direction), preventing shear-out by providing enough edge distance, and ensuring that the joint is safe in both bearing and net-tension, but would fail in bearing first by providing adequate net-section width. Tests done by various investigators[58,59,68–71] have shown that $e/d \geq 3$ and $w/d \geq 5$ with $t/d \geq 0.6$ allows adequate bearing strength to be developed. It is well established and widely recognised that the near-quasi-isotropic lay-ups provide an optimum joint performance. However, the stiffness requirements often necessitate more plies in the load direction. Various studies[40,44,49,60] have shown that the lay-ups be limited to having at least 10% plies in every chosen orientation and about $^{1}/_{8}$ to ½ plies of 0°, $^{1}/_{8}$ to $^{3}/_{8}$ plies of 90°, and ¼ to ¾ plies of ±45°. A typical lay-up having 0°/+45°/–45°/90° plies in the proportion of 5/2/2/1 thus provides a realistic worst case and is often used for allowable bearing data generation. Bearing strengths for various lay-ups obtained in one study[71] are shown in Figure 4.

One of the significant aspects in deciding the bearing allowable is about defining the bearing failure. It is to be realised that depending upon the laminate configuration, significant amount of hole elongation may occur before the final failure stress is reached. Such hole elongation may cause hammering at bolt-hole interface in cyclic loading with stress reversals.[49] There has been some debate

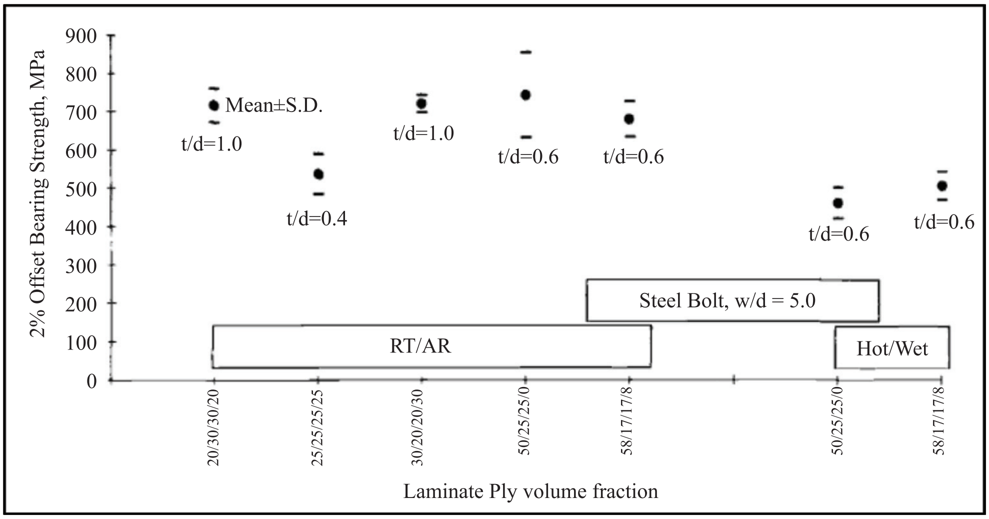

Figure 4: Bearing strengths for various laminate configurations and effect of hot-wet.[71]

on how much hole elongation can be acceptable and set as criterion for defining the bearing strength. Two criteria have been put forward and have gained some acceptance in the industry: 2% elongation in case of the earlier generation of epoxy resins which are relatively brittle (which is used in the study[71]), and 4% elongation for the newer, toughened resins.[82]

4.5 *Considerations for fastener joints*

In addition to selection of allowable strengths for laminates with holes and for bearing, there are a few significant aspects which need to be considered for proper use of allowables when designing fastener joints. The failure modes of a fastener joints are shown in Figure 5. As mentioned earlier, the geometric parameters (edge distance, width, pitch etc) are to be so chosen so as to allow full bearing strength to be developed. In addition, following aspects are important.

4.5.1 Effect of clamp-up pressure or lateral constraint: It is known that the lateral constraint provided to the laminate plies helps in preventing or delaying the micro-buckling of fibres and thus result in higher bearing and open hole compression strengths.[39,42,44,59] Such lateral constraint may be available through clamp-up pressure in a torqued bolted joint or may be provided by other plies in a thick laminate.[44,59,72,73,85] Bearing strength increase from 20% to 100% has been reported for various types of laminates. However, it is to be noted that creep and relaxation in laminates over a long period, as well as vibrations, can reduce the effective lateral constraint over a long time and thus this advantage may not be available in practice. A study on relaxation of bolt-clamp-up[48] has shown that reduction of the clamping force can be 20% in a year and 32% over 20 years. A prudent way to handle this is to generate the allowable strengths from tests on finger-tight joints and not on torqued bolts[70,71,76] or to use a knock-down factor (of up to 2) if the data is generated with torqued bolts. Similarly, it is prudent not to use thick laminates for allowable generation; usually, thickness up to and close to the bolt diameter is considered satisfactory. Further research in understanding the 3-D behaviour in clamping through FEM models is reported in.[80,81]

4.5.2 Bearing-bypass interaction: It is to be recognised that in a realistic structure, often only a part of the load is reacted at the hole and the rest is “bypassed” to be reacted at some other constraint. In particular, in a multi-bolt joint it is important to assess bolt loads at each bolt and use proper strength allowables for checking for bearing and net-section failure. Methods suggested by Hart-Smith,[38,46,47] Lafon[57] and those developed by Crews and Naik[61] can be used to make such assessment. One scheme to arrive at allowables is to use the knocked down laminate tension strength for net-section failure and use bearing allowable on the bolt-load.

4.5.3 Countersunk holes: While most of allowable data is generated on full cylindrical holes, knock-down factors need to be applied when countersunk holes are used. A study by West[56]

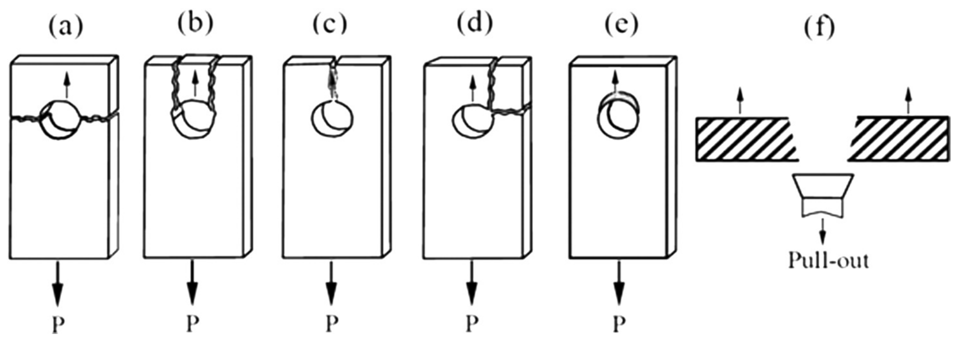

Figure 5: Failure modes of fastener joints: (a) Net tension, (b) Shear-out, (c) Transverse splitting, (d) Cleavage, (e) Bearing, (f) Pull-out.

shows that laminates with higher bearing strength also show more reduction due to countersinking. About 25% reduction is found to be common when sufficient thickness is provided (say, $t/d \geq 1.3$). However, for smaller thicknesses, even higher reduction needs to be provided for.

4.5.4 Bolt pull-through: The bolt pull-through failure (Fig. 5(f)) can become critical in thin laminates, especially when countersunk fasteners are used. In absence of any bending, it is a simple exercise to calculate bolt pull-through load on the basis of the allowable transverse shear stress for the laminate and the area being sheared (i.e., thickness times the bolt-head circumference). One study on skin-rib joints[63] suggests that this is conservative and the bolt pull-through can sustain as much as 1.5 times this load. However, when bending is present (which is the realistic case) the pull-through can occur at half of the loads.

4.5.5 Single shear (lap) joints: Most of the test data for bearing strength is usually generated on double shear configuration. It is generally recognised that the single shear (single lap) joints exhibit reduction in strength due to two factors: one, rotation and bending of the bolt and the other, bending of the laminate. In a well-designed joint with large overlap lengths, the effect of laminate bending is not significant but there can be 20–25% decrease due to bolt rotation and bending.[58,60,71]

4.5.6 Pitch in multiple fastener joints: Hart-Smith[40] has shown that for a fastener in a multiple fastener configuration with pitch p, the stress concentration is less than that for a single fastener in a strip of width equalling pitch. Thus, data on single bolt tests can be applied to multiple bolts. Matthews[59,70] also suggests that data on single bolt with $w/d \geq 4$ can be applied to $p/d \geq 4$.

5 Impact Damage

5.1 *Impact damage behaviour*

It is well recognised that due to poor strength in normal-to-plane direction, laminated composites are susceptible to delamination and damage due to impact. The impact damage may involve delamination, matrix cracking, and even fibre breakage. This severely restricts the residual strength, especially in compression. A major concern is that the impact damage may remain invisible or undetected. While designing, an impact threat scenario is generally considered for the structure and a design philosophy is drawn up which defines the impact levels that the structure should sustain along with inspection and maintenance intervals, in consultation with the regulatory or certification authorities. Guidelines for such an exercise have evolved over the years and continue to do so. A good guideline in this connection is provided in a study by the US office of aviation research.[107]

Extensive studies have been done to understand the damage due to impact and useful reviews on this can be found in Chap 7 of MIL-HDBK-17-F Vol 1 and 3[13,15] as well as.[108–112] A low velocity impact generally incites an overall geometry response in the structure while a high velocity impact (such as by a projectile) brings about only a localised mode of deformation and energy dissipation over a small area. Also, at low velocities, a flexible structure can absorb energy in flexing and in flexural failures of fibres, while for a stiff structure energy gets absorbed in interlaminar shear and consequent delamination. On impact, compressive stress waves generated at the point of impact travel through the thickness and are reflected as tensile stress waves from the back surface. Thus, most of the damage gets initiated at the back face, causing a cone of internal damage. At very high velocities, a projectile can shear out of the laminated structure causing full or partial penetration. Thus, for generation of design allowables, it is important to note that tests for low velocity impact should be designed with target geometries representative of the desired structure while far field geometric effects are not very significant for high velocity impact.

Among the common impact threats that are considered for design allowable generation are tool (including some service boxes or equipment) drops, runway debris thrown by tires, hail-stones and some bullet strikes. The two major issues to be addressed through proper tests are: one, the detectibility threshold (for visual inspection and for other NDE) and the other, the loss in strength. In most cases of the current design practice, the design philosophy has been built around the concept of "Barely Visible Impact Damage" (BVID). Any damage which cannot be seen (BVID or less) must be tolerated by the structure (with Design Ultimate Load) through the life time (or any other specified period). However, for thick laminates and laminates with impact resistant matrices, the impact energies to cause BVID can be very large and the threat of such high impact may be improbable. An energy cut-off is therefore defined for the structure to sustain, see Figure 6(a). Thus, a common practice is to base the design on the Visibility cut-off and the Energy cut-off.[15,107,113] In terms of damage sustenance, apart from the BVID, Allowable Damage Limit (ADL) and Critical Damage Threshold (CDT) are generally specified which need to sustain the design ultimate load and the design limit load respectively [Ref.[15] Chap 7]. See Figure 6(b).

5.2 *Compression after impact as design allowable*

By far, the low-velocity blunt object impact is considered to be the driving case for arriving at design allowables, as it can leave a large damage undetected. The compressive strength is the most affected and thus Compression-After-Impact (CAI) strength has become an important parameter for designers. There is a complex relationship between the extent of damage, impact energy, impactor shape and mass, laminate thickness, lay-up etc. and one needs to judiciously select test parameters to represent the actual design geometries. A good review of the literature on these can be found in.[114] Effects of lay-up and stacking sequence, particularly from the view point of taking advantage of 0° plies, have been reported in[115–118] and a good discussion of this can be found in.[117] Effect of impactor shape are studied and reviewed in[119,120] and that of impactor mass in.[121] Effect of plate thickness and impact parameters is studied in[122,123] for woven fabric laminates. A sensitivity analysis of various parameters affecting the impact resistance of laminates is given in.[124] From the mechanics, one can observe that in thin laminates there is significant flexing and thus energy for damage initiation increases with thickness while in thick laminates where interlaminar shear is the dominant mode, it is proportional to the reciprocal of thickness.[125–126] Over the years, the industry has

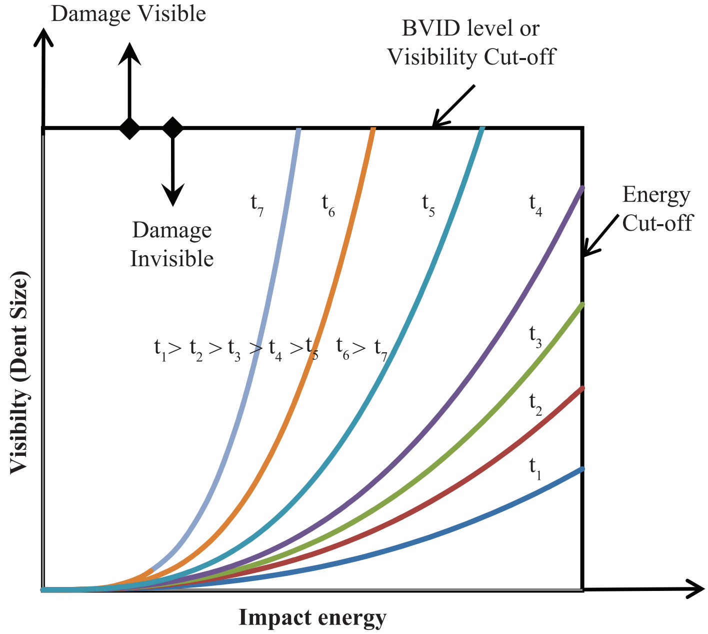

Figure 6: Visibility and energy cut-off for Impact damage with laminate thickness t.

created and used several "standard" fixtures and tests to determine the CAI for their requirements; Boeing and NASA fixtures being more common amongst them, i.e., Boeing BSS 7260[127] and NASA RP 1092.[128] The methodologies have evolved over the years leading to NASA RP 1142,[129] SACMA SRM Method,[130] and more recently, ASTM standards ASTM D7136 and D7137.[131–132]

A study made by NASA[133] on several composite materials showed that failure strains (in compression) of laminates are reduced by about 70% by impact damage to about 3000 microstrains for impact energies of about 30 ft-lbs (~40J). Other studies using different configurations by Cantwell and co-workers[134,135] have also shown that the strength reductions of about 55% can be expected due to impact damage. However, the impact energy required to cause BVID can be much less. Studies by Cantwell et al[134] and Bishop and Dorey[136] have shown that for thin laminates (thickness ~2 mm) the BVID energy is around 2–3 J. Other studies[137] have also shown that the BVID energy can be expected to be 0.6–1.5 J per mm of laminate thickness for laminates of thickness up to 4 mm. Even though the impact energy required to cause BVID is not strictly linear with the laminate thickness, this is found to be a good parameter to work with in preliminary design and for planning of tests. Exploring the damage caused by sharp tools such as screw driver as against that caused by blunt objects, Geier et al[138] have shown that the damage caused by falling hammer is already visible and the strength reduction is similar to that caused by a 6 mm hole. These experiments also showed that the impact damage in wet condition is actually less than that in dry condition. This indicated that for impact damage tests, the impacting should be done on non-aged samples which can be later aged and tested to find the hot-wet CAI. Several studies such as[43,45,55,137–146] using different types of tests have shown that for carbon epoxy composites using standard modulus carbon fibre, the allowable failure strains based on compression after impact with BVID damage can be around 3900–4200 microstrains. Results from one such study[145] are shown in Figure 7. Similar studies[134,135,144] have shown that, for tension after impact the failure strains are higher than those in compression and are about 4500 microstrains.

5.3 *Detectibilty of damage*

The issue of what is "barely visible" and the visibility cut-off has received significant attention over the years. As reported by,[140] the US MIL specifications in 1980's set the dent depth 2.5 mm as criterion for BVID. Other regulatory agencies elsewhere, and in particular for civil aircraft, have set the visibility criteria to be around 1 mm dent depth or left it to be defined by the method of inspection used. For example, para 5.8 ACJ 25.603[147] states that, "It should be shown that impact damage that can be realistically expected from manufacturing and service, but not more than the established threshold of detectibility for the selected inspection procedure, will not reduce the structural strength below ultimate load capability". Investigations on composite laminates have not been always in support of a strong correlation between the dent-dept and the internal damage. For example, the study in[148] show that the dent depth does not correlate well with internal damage in case of thin (<2.4 mm) laminates. Also, there is some amount of relaxation of the dent and this also needs to be accounted for.

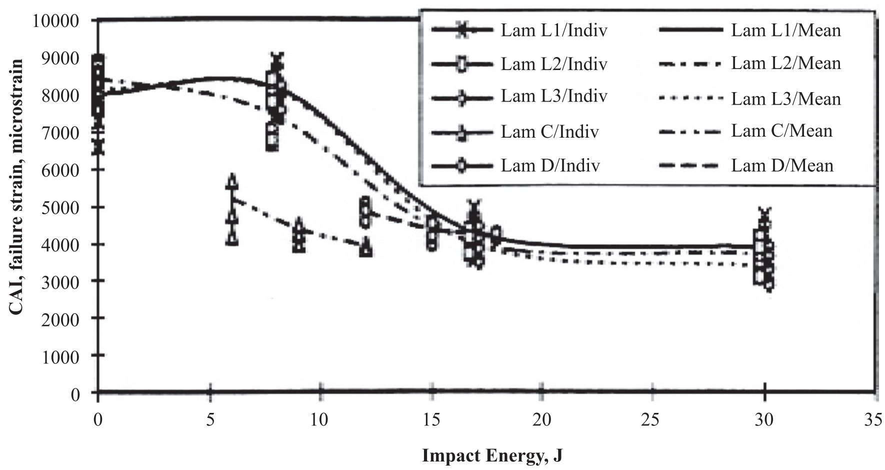

Figure 7: Compression after impact. Failure strains for quasi-isotropic laminates. Laminates L1, L2, L3 are 6 mm thick with various stacking sequence. Laminate C is 3.6 mm thick, Ref.[145]

Nevertheless, dent depth has come out as a useful marker of impact damage. In practice, dent depths of 0.25 mm to 1.0 mm have been used variously as criteria for visibility depending upon the distance of visual inspection (or other specified inspection method). Boeing, for example, has typically used 0.25 mm–0.5 mm dent depth to be visible from distance of 5 feet in typical lighting condition as BVID condition.[149] A comprehensive study and discussion of various aspects can be found in a recent paper by Cook et al[150] which concludes that detection rates are affected by flaw depth and flaw width, surface colour and finish, and environment lighting. It suggests that flaw size limits should be based on visual tests for worst case (matt blue) samples and that it is possible to improve the effectiveness of inspection using specific lighting arrangements (e.g. grids) and surface paint colours/finishes. There is also some effort to establish the correlation of dent-depth to internal damage by analysis. For example, a modelling approach is outlined in Ref.[151] to predict the permanent indentation due to impact.

5.4 *Impact on preloaded laminates*

Earlier studies such as by Geier et al[138] had not shown any significant effect of preload (while impacting) on the damage size. However, a more recent study[152] has indicated that such preloading can increase the damage size when impacted. In another study[153] which uses also impact response modelling, it has been seen that the effect of in-plane preload diminishes at higher impact energies. Effect of compressive preload is studied in[154] through simulation and experiments which showed increase in deflection and energy absorption, but the effect was not very pronounced. A study based on FE simulations[155] reports that the span-to-thickness ratio is a fundamental parameter in determining the effect of preload. Under a tensile preload, the peak stresses caused by impact were found to be higher than in the case of no preload. Under compression, the most significant influence of initial stresses was found at medium span-to-thickness ratios for preloads comparable with the buckling load. In other cases, negligible or even beneficial effects were observed. The study done by DLR Germany[156] also found that while the tension preload did not have much effect, the compression preload is the most critical case for blunt impact and delaminations grew quasi-statically for preloads half of the buckling load. Modelling the dynamics of impact, the study in[157] found that preload can actually raise the CAI strength if the load approaches the initial buckling value. However, as the preload approaches the CAI strength the induced delamination can propagate catastrophically during the impact. On the other hand, the experimental results of[158] showed that both, pre-tension and pre-compression, influenced the impact behaviour and that the pretension may induce the severe effect for impacted composite laminates. The experiments of[159] also showed the influence of compressive preload was to reduce impact tolerance to some extent. Nevertheless, the current standard test practice is not to pre-load the test specimen for generating CAI data.

6 Discussion and Future Developments

6.1 *Material systems*

For the first generation carbon/epoxy composite primary structures, the design driving factors have been the open hole strength in tension and the post impact strength in compression. The overall allowable design strains have hovered around 4000 microstrains. Imparting toughness and improving the impact damage resistance and tolerance appears to be the key to raise the allowable strains to a higher value so as to realise the full potential of composites. Use of higher grades of carbon fibres (larger failure strain, more strength) with toughened matrices is one obvious way and which has been and continues to be pursued vigorously. It is well established[160] that tougher resins will improve the damage resistance and tolerance so that CAI can be better. Currently, intermediate modulus fibers with toughened epoxy or bismaleimide or other resins which have been researched since late 80's[161] are commercially available (see for example,[162]) and most of the current aircraft development use those. Later and current approaches include use of nano-materials: using nano-clay and carbon nanotubes as reinforcements into the matrix or as coatings on carbon fibers[163–166] and use of glass or aramid fibers and fabrics along with carbon fibers to create hybrid composites.[167–168] It would appear that one may sacrifice the strength in neat condition if some advantage can be gained in damage resistance and tolerance so that the ultimate design allowable values are increased. This seems to have also inspired use of through thickness reinforcement, such as stitching and z-pinning[169–171] to improve damage tolerance and thus improve the allowables. There is also an innovative approach to use self-healing materials, and also combine it with through-thickness reinforcements so that some damage is healed and overall damage is contained.[172–173] Yet another aspect of improving materials is to reduce the hot-wet effect and this may bring in new materials. With such newer materials, understanding the behaviour of the material with their new features will be the key to set up design allowables even as the process to establish design allowables would be on similar lines.

At the present stressing level of 4000 microstrains, fatigue is not a critical issue in the sense that the laminated composites under in-plane loads have higher endurance limit and that the damage progression under in-plane loads is slow enough to satisfy fatigue life of million cycles. With the expected increase in allowable strains, whether the fatigue issues will become critical is an open issue. Investigations on these lines are of current and future interest as seen from some of the research reported.[174–176]

6.2 *Delamination assessment through fracture mechanics*

Apart from the impact damage, another consequence of the poor out-of-plane tensile resistance (peel resistance) of the laminated composites is the delamination of plies. The semi-monocoque construction of aircraft structures mostly ensures that the composite structure is subjected to in-plane loads; however, there are instances of three dimensional states of stress which cause peel loads and one needs to ensure integrity of the structure under such conditions. The structural features which are important for such examination are: ply-drop areas, T-joints (either co-cured or co-bonded, bonded) and delaminated sub-laminates either due to impact or as manufacturing defect. In general, these are taken care of by proper design, i.e., proper geometry and provision of load paths. Thus, they do not directly form the subject matter for design allowable values. On the other hand, developments in fracture mechanics of composites in last couple of decades have made it possible to assess propensity to delaminate and also propensity of the delamination to grow once formed. Fracture mechanical parameters of delamination toughness represented by such parameters as Critical Strain Energy Release Rates (SERR, denoted by G) particularly in the opening mode (G_{Ic}, Mode I) and sliding shear mode (G_{IIc}, Mode II) and the corresponding fatigue thresholds (G_{Ith} and G_{IIth}) then enter into the assessment. The research done over the years towards finding suitable measure for delamination toughness, has led to ASTM standards for Mode I and good progress has been made for standard for Mode II.[177–181] Concurrently, the FEM analysis of structures along with the Virtual Crack Closure Technique (VCCT) and modified VCCT (MVCCT) have enabled determination of SERR at delamination tips.[182–184] Application of these techniques, particularly where delamination growth is expected to be self-similar due to the constrained geometry has been found to be quite useful in assessment of certain cases of delaminations, see for example, application to ply-drop problems,[185–186] to T-joint problems[186] and to delamination growth due to sub-laminate buckling.[186–189] A sample result from[186] for a delamination in the shoulder of a T-joint and skin-stiffener connection is shown in Figure 8.

6.3 *Use of modelling and simulation*

With increasing availability of computational power, there has been a significant emphasis in research community to develop models for progression of failure under various kinds of loading (tension, compression, impact, etc) for the typical configurations such as, open and filled holes, T-joints, stiffened plates, etc. (See for example[80–81,83–84,88–90,96,98–103,106]) and with newer features of materials as mentioned in Sec 6.1. Such efforts are expected to aid in removing some of the ignorance factors used to knock down allowable values. As noted above, the holes and the impact damage are the two most limiting features, it is not surprising that these have attracted the attention of the research community. Some of the efforts are briefly stated below as a sampling of the research interest. Much of this research is aimed at creating progressive damage models which can then be used to predict the strength or other parameters of interest.

Bearing behaviour is studied in a Ph.D. Thesis[79] with emphasis on temperature effects. References[80,81] give a 3-D FEM model to study effect of clamping loads on bearing failure and bearing strength. In[83] and,[90] a Failure Area Index is developed which is used for failure prediction. The method is shown to work within 20% of the experimental results. The work in[84] deals with thick laminates and develops modelling technique for deformation analysis. Work in[86,87] develops an analytical tool which can give detailed information, complementary to the experimental data, on matrix cracking and fiber breakage onset and growth in composite single-lap joints with different bolt types and sizes. Work in the two Ph.D. theses[88,89] deal with stresses in multi-fastener joints. Using elastic analysis and FEM they derive formulae for stress concentrations in multiple hole joints considering interactions such as secondary bending, contact, etc and build models leading to prediction of bearing strength and joint strength. In,[92] the joint geometry effect on the fracture mechanisms is analysed and a failure map is obtained, identifying three regions of typical failure modes of mechanically fastened joints.

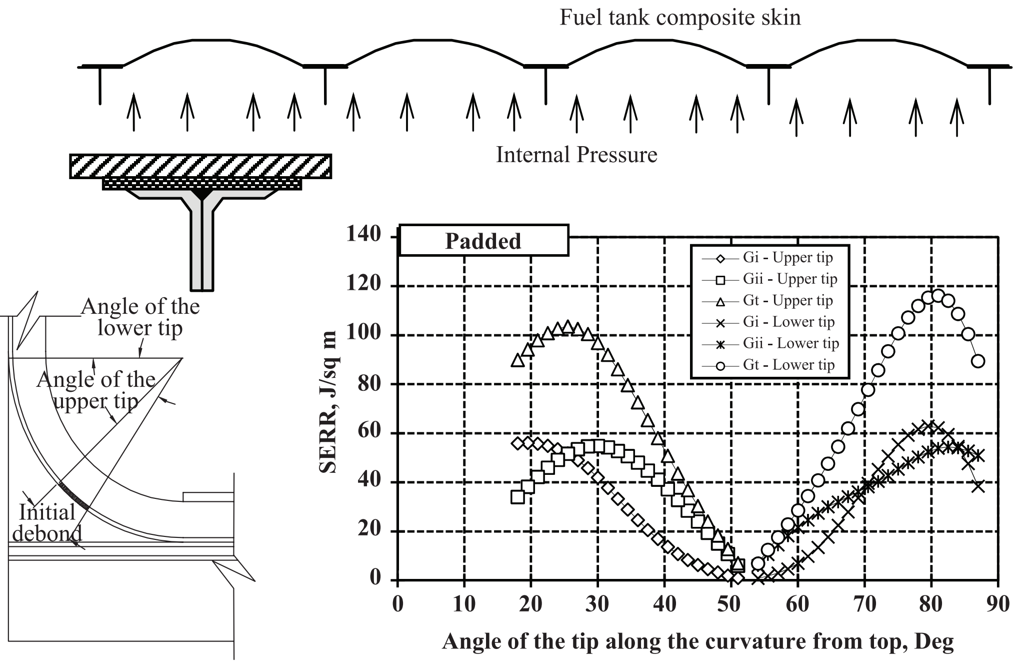

Figure 8: Strain energy release rate for a delamination in the shoulder radius of a T-joint in skin-stiffener connection. The deformation under internal pressure causes peel stresses on the delamination. The criticality of a size of delamination can be assessed by comparing max SERR (G_I, G_{II}) to its critical value (G_{Ic}, G_{IIc}). Source Ref.[189]

Reference[99] uses limit analysis methods to predict joint collapse load. Rosales et al[100] consider various combinations of failure theories and degradation models, as well as various ratios of bearing/bypass loads. Reference[101] uses cohesive zone model to predict open hole compressive strength of a toughened carbon/epoxy quasi-isotropic multi-directional laminate and investigates the level-ply scaling or ply blocking effect on notch sensitivity. Reference[102] proposes new failure criteria for joints based on 3-D state of stress and accounting for nonlinear shear stress-strain behaviour. In[103] a model capable of direct simulation of failure of composites containing open holes is presented which correlates well with experimental data for matrix and delamination crack growth in graphite-epoxy quasi-isotropic composites with open hole and with thick plies, where the composite fails in the delamination failure mode. To complement the theoretical models and to generate test data, there is wide range of exploration through experiments on newer tougher materials similar to the ones seen earlier for first generation of carbon-epoxy. For example, Ref.[91] reports experiments over a wide range of temperature, studying evolution of microscopic damage and compressive kinking of 0° layers having a major influence on bearing strength. Experiments using single fastener in double shear are reported in Ref.[105] to study subcritical damage mechanisms of bolted joints which corroborates use of cohesive zone elements for such a study. Hot-wet effect on open hole compression for a tougher material system is studied in[94] from the view point of applying hot-wet considerations of aircraft to space craft. A recent paper[106] gives a comprehensive study on effect of thickness and laminate taper on the stiffness, strength and secondary bending of single-lap, single-bolt countersunk composite joints and argues that significant weight savings can be obtained by using tapered laminates. Ref.[104] consolidates the understanding on joints behaviour into giving a design and analysis guide on structural joints.

Low velocity impact behaviour and damage creation is modelled using FEM in Ref.[153,190–193] Ref.[153] also includes damping effects. In,[157] the coupling between impact and preload is simulated using the equations of motion. Reference[155] presents a study on the effect of preloading through finite-element analysis of several impact events on laminates with different span-to-thickness ratios, tensile and compressive preloads, both uniaxial and biaxial. Use of commercial software such as LS-DYNA for simulation of low velocity impact with compressive preload is reported in.[154] A modelling approach using a damage index parameter to investigate damage growth and accumulation under repeated impact is outlined in.[194] Simulation of impact events is used to study strain-rate effects for toughened composites in,[195] whereas a possibility of increasing CAI by exploring non-symmetric laminates is explored in.[196]

7 Concluding Remarks

Choice of design allowable strengths for composite materials has an important bearing on the entire design process of an aircraft and the final product and its certification and performance. We have outlined some important considerations while arriving at design allowable values. In doing so, we have examined the effort that has gone into developing underlying understanding of composite behaviour and its implication for the allowable values. In particular, we looked into the hot-wet effect, holes and fasteners and impact damage. The literature on these issues is really vast and the references quoted here should be taken as only representative of the entire effort. No claim is made to being exhaustive, nor is any claim made towards relative importance of cited literature over the omitted ones.

At the present design allowable values, the potential of composites as aircraft structural materials remains vastly unrealised. Future efforts need to be and will be directed towards expanding the current envelope of usage, such as for example, exploring post-buckling strengths, strength beyond first-ply-failure, use of non-symmetric lay-ups and through-thickness reinforcements. For, this to happen, it is important to develop better understanding of the material behaviour in those regimes and develop better predictive capability. Better material systems would be explored and toughness and resistance to hot-wet would be the target directions. Newer developments such as Structural Health monitoring can bring in new thinking about allowable damage which also will hopefully allow higher design allowables and better utilisation of the potential of composites.

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Much of my understanding and knowledge of designing with composites is derived from the work carried out for the LCA programme during 1980's and 1990's and this forms the cornerstone of the discussions in this paper reflecting a view from a practising engineer rather than a research scientist. Many colleagues and collaborators, too numerous to name, from ADA, HAL, NAL, IISc, IITs and other collaborating institutions have helped me in learning about composites and their contribution is gratefully acknowledged. In particular, thanks are due to Dr. KN Raju, head of the erstwhile National Team for Composite Wing, and my colleague at ADA for long time, Dr. K Vijayaraju. Thanks are also due to my teacher at IISc, Prof B Dattaguru. The help and guidance received from Dr. AR Upadhya, Dr. Kota Harinarayana, Shri DY Katti, Shri TGA Simha and Prof KP Rao through various discussions from time to time is gratefully acknowledged.

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