IS THERE REALLY NO NEED TO BE ABLE TO PREDICT MATRIX FAILURES IN FIBRE-POLYMER COMPOSITE STRUCTURES?
PART 2: EXAMPLES OF MATRIX FAILURES PRECEDING FIBRE FAILURES

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ABSTRACT
This paper describes typical situations in composite structures where non-catastrophic matrix failures precede fibre failures, but which all of the most popular composite failure theories, based on artificially homogenised laminae, are inherently incapable of predicting. This emphasises the very real need for improved theories advocated in the companion paper on the theory and related physics. Examples are selected from both secondary and primary structures. One chronic problem, arising in many guises, is the excessive clustering of too many parallel fibres for the polymer resin in which they are embedded to transfer loads in and out of them. Another common problem is abrupt interruptions of load paths, with only the matrix to diffuse the end loads. The interruptions can be the result of both poor design practice and impact damage. Some matrix cracks shown cannot be associated with pre-existing starter cracks, thereby devaluing theories requiring their pre-existence in order to apply the theories. Worse, the absence of theories with which to characterise the adverse consequences of an inadequate resin content have led to conscious efforts to minimise resin content, to maximise fibre-dominated longitudinal lamina strengths, with no awareness of the consequential reduction in load carrying capability when the matrix is dominant.

Keywords: fibre-polymer composites; matrix failures; illustrative examples

INTRODUCTION
Previous efforts to improve upon the design and analysis capability for fibre-polymer composites have focused on identifying analytical deficiencies in the existing theories and proposing new theories to overcome them, or to expand on the features that can be included in the process. These attempts have been successfully rebuffed by the defenders of the status quo, who have strenuously opposed all such attempts that introduce revolutionary changes, while accepting only evolutionary changes that did nothing to eradicate the inherent problems.

The emphasis of Part 2 of this paper is therefore a series of actual practical situations in which matrix failures precede fibre failures in composite laminates, thereby preventing the fibres from achieving their intrinsic strength, which is what should be predicted by any theory that omitted consideration of distinct matrix and fibre constituents. None of these matrix failures can be predicted by the established theories the author has condemned, yet they are typical of very real problems that have occurred on actual structures. The reason for this different approach is to educate the users of the many bogus composite failure theories as to what might have caused all the unanticipated premature failures uncovered belatedly by testing, in the hope that the
customers who suffered the most might have a sufficiently powerful voice to be able to demand something better to work with next time.

The examples selected are each typical of a whole class of similar problems. The first problem illustrated is the excessive clustering of too many parallel fibres for the resin matrix, in which they are embedded, to transfer load in and out of them. This situation is most commonly caused by poor design practices, many of which are lauded as being good ideas and actively encouraged, because there has been no analytical capability to warn of the consequences, but can also arise as the result of damage during manufacture or in service. Fibres need to be thoroughly interspersed, not clumped together. Also, fibre tow sizes need to be limited. Another class of problems is associated with distortions created by resin shrinkage whenever there is an excessive local concentration of resin. Next, the effect of stacking sequence on edge delaminations and internal delaminations caused by impact is described. Again, it is shown that interspersal of thin plies is the way to go and that thick plies in the name of false manufacturing cost reductions is an invitation to the creation of such problems. Another popular composites design concept with under appreciated consequences is the use of 0° rich hard strips to act as periodic crack stoppers between spans of much softer laminates. They stop cracks spreading from the soft regions, but by turning them to run unrestrained along the length of the crack-stoppers. This secondary failure is not predicted by homogenised theories. The consequence of poorly designed stiffener run-outs is also illustrated. Again, the end load at which they start delaminating requires an analysis of the matrix interface between the stiffener and the skin. This problem has recurred in many applications over many years; it is most frequently caused by a desire to decrease part count, without regard for the consequences, by co-curing composite structures in the false belief that they were equivalent to integrally stiffened metallic structures (which tend to suffer fatigue cracks at the ends of the stiffeners whenever they stop short of the edge of a skin panel). Eccentricities in load paths, and the direct application of peel loads, can also cause delaminations, of which the initiation cannot be predicted by any of the theories the author has criticised because they fail to differentiate between the fibre and matrix constituents. The illustrations include good design practices for avoiding some of the potential pitfalls associated with premature matrix failures. Some examples are included to show how delaminations can occur without the pre-existence of starter cracks. Obviously, none of the theories requiring the presence of an existing crack or delamination, before they can even be applied, are appropriate in such cases. But SIFT is! Finally, the subject of delaminations caused by forcing together detail parts that don’t quite fit is raised. In the absence of an ability to predict matrix failures, it is not possible to rationally establish appropriate manufacturing tolerances, which have been found by costly experience to be far less that the equivalent tolerances established for truly homogeneous (metallic) structures.

Hopefully, these examples, and the explanations of why the matrix failures occurred, will encourage those people who have suffered from such problems in the past to first understand why they did and then to demand something better sufficiently vociferously to overcome the past resistance to allowing such improvements to be developed and applied whenever the proposed improvement invalidated the prior work.

**PROBLEMS ENCOUNTERED WITH THE INCLUSION OF BLOBS OF UNIDIRECTIONAL REINFORCEMENT IN COMPOSITE STIFFENERS**

One real-world problem caused by the clustering together of too many parallel fibres is the fashionable noodle inserted in the corners of stiffener cross sections, shown in Figure 1.
Sometimes, this is aggravated by the inclusion of undispersed reinforcements in the flanges, to boost the longitudinal and bending stiffnesses of the stiffeners. A related concept is the addition of pre-cured all-0° reinforcing rods in flanges to enhance their longitudinal capability.

While the rods are superior to the slabs of longitudinal fibres sometimes added to flanges, because of a far better interfacial perimeter-to-cross section ratio, they both share one major drawback; they cannot be repaired if they are broken. Neither can the noodles at the intersection of webs and flanges. If they are too large, they will progressively detach themselves from the rest of the structure as the result of thermal cycling (or mechanical loads), which is explained below.

Regardless of whether or not these are sound structural concepts in practice as well as theory, a model that differentiated between the fibre and resin matrix constituents would be able to predict that there is an upper size limit on the concept, for the reasons explained in Figure 13 of Part 1 of this work. Any theory in which each layer was homogenised, with all interfaces between layers thereby eliminated, would be incapable of doing so. So, there would be no warning that there was no load path to get load in or out of the ends of such localised reinforcement.

The irony about the use of load-carrying noodles is that any increase in longitudinal strength is associated with a reduction in pull-off strength for the stiffeners. It seems counter-intuitive, but secondarily bonded back-to-back angles, with nothing but air in the interstice, are found by test to have a greater pull-off resistance than co-cured angles with either fibre or adhesive fillers in the interstice. This is explained in Figure 2.
Despite the air gap under the stiffener web in the diagram on the left, versus the direct load path through the noodle in the diagram on the right, the left-hand design is found by test to be consistently stronger. The reason is that the radial thermal contraction of the noodle is transmitted to the adjacent layers in the skin. This consumes some of the available interlaminar tension strength, while the direct load path adds even more pull-off load to this prestressed area. This is easy to comprehend when the fibres and resin constituents are properly segregated, but impossible to predict using a homogenised material model.

Experienced designers and analysts of fibrous composite structures have learned to apply empirical design constraints to prevent many of these premature matrix failures, essentially by limiting the thickness of fibres adjacent to each change in fibre direction in the layup, but sometimes they are overlooked. The running shear loads along the length are inevitably very much smaller and do not exceed the capabilities of the matrix interface adjacent to the fibres. But these concentrated reinforcements share the characteristics of adhesively bonded joints in which, with the exception of linearly tapered scarf joints, almost the entire load is transferred in very narrow zones adjacent to each end of the overlap. This phenomenon is depicted in Figure 3. Any peel stresses induced by local or global eccentricities in the load path are even more concentrated, and peak at at least one end. If the adherends in Figure 3 are too thick, and the ends not tapered to alleviate the severity of the load transfer, the adhesive, or resin, will fail once sufficient load has been applied, before the fibres are fully loaded.

As an aside, it should be noted that, if the adhesive layer had been omitted from the bonded joint model in Figure 3, the high, but finite, peak shear and peel stresses would be replaced, analytically, by singularities. Most people involved in this bonded joint work know better than to do so but, when it comes to edge delaminations, which are discussed here later, most of those people assume that the correct model is one of homogeneous layers butting up against each other, with zero-thickness interfaces in between. This is how they generate the “singularities.” Otherwise, the fracture-mechanics type of analyses could not even be applied.
The problems with concentrated $0^\circ$ reinforcement of composite laminates are not confined to mechanically applied loads. If the ratio of cross-section of reinforcement to perimeter of effective interface is high enough, there can be severe interfacial loads induced near the ends of the stiffener caused by thermal mismatch effects. While the coefficient of thermal expansion of typical multidirectional carbon-epoxy laminates is very much lower than that of aluminium alloys, for example, the longitudinal coefficient of thermal expansion of the $0^\circ$ reinforcement is essentially zero. Consequently, when the laminate cools down after curing at high temperature, the basic portion shrinks in all directions, but the shrinkage is resisted in the longitudinal direction by the concentrated $0^\circ$ reinforcement. This develops intense shear stresses in the matrix surrounding the reinforcement at each end of the composite component, as shown in Figure 4. If the cross-sectional area of this concentrated reinforcement is too high, the matrix will actually fail, under shear loads, creating delaminations that could, theoretically, in time, progress all the way along the length, until they meet in the middle. The “ends” of the noodle would be continuously redefined. Actually, the initial delamination would be likely to move away from the interface into the basic laminate, where the crack front is more diffuse and fibres crossing the crack surface retard its further growth. If the cross section of reinforcement is not quite that high, the application of external mechanical load will also induce high shear loads in the same matrix interface at each end of the member. These two peak shear stress contributions will combine at one end of the member and tend to cancel each other out at the other end, as shown in Figure 4. Ironically, it can be the compressive mechanical load that causes the delamination – which is something that is certainly not expected and which is almost impossible to explain scientifically if one denies the heterogeneity of the fibre-polymer constituents of what are most frequently wrongly considered to be anisotropic homogeneous materials.
One reason cited for this preference to add local longitudinal reinforcement to composite stiffeners is the myth that it is desirable to “tailor” the fibre dispersions to show how superior composites are to metals, because it is not possible to do the same for the latter. This technique is projected to reduce weight. Done in moderation, with the additional fibres dispersed in small cross sections so as not to exceed the capabilities of the resin matrix, there are some situations when this can be beneficial. But the so-called noodle is not one of them, even for thin laminates, as is explained below. However, there is another attraction associated with the noodle, which comes with a hidden trap, which was explained in Figure 2. It has been shown by testing for the LearFan all-composite aircraft that the highest stiffener pull-off strengths are attained using secondarily bonded pre-cured stiffener and skin components with a moderate radius in the corners – and a gap underneath the stiffener web. This is because, then, the fibres are essentially wrinkle-free. Unfortunately, this gap inhibits the use of ultrasonic inspection of the stiffener, from the outside of the skin, to detect internal delaminations. So, there has been a desire to fill the cavity. The best such filler is made from liquorice sticks of adhesive film. These have a low modulus, do not pick up any significant load, and do not create delaminations when they are too large. But they add weight, and the desire of some designers to improve their composite structures by filling the cavity with load-carrying material is often irresistible. If they overdo it, they end up with real delaminations, which prevent further ultrasonic inspections deeper down in the stiffener but, worse, which are unrepairable. This potential, and sometimes real, problem will persist as long as the pretence that fibre-polymer composite layers are homogeneous is maintained.

There is another very costly associated problem with filled noodles, whether by bundles of 0° fibres or by liquorice sticks of adhesive film, which shows up particularly with thin-faced composite sandwich panels as are used on aircraft fairings and cowlings. This is purely an economic issue with co-cured designs; the stresses are usually low enough to tolerate the associated wrinkled fibres, at least in secondary structures. This second problem is one of resin shrinkage, as depicted in Figure 5.
The concentrated blobs of reinforced, or unreinforced, resin (or adhesive) at the corners of composite stiffeners that are co-cured with the skin cause highly visible shrink lines on the exterior surface. Whether or not they are structurally tolerable, they have to be filled by hand glazing to improve the appearance of the external surface – yet another additional step in this one-shot processing technique before the panels can be primed and painted. The problem is so acute that one manufacturer of engine cowls pre-cures the exterior skin in a separate step, because it was established there that this was far less expensive than curing the structure in a single step and having to glaze it all by hand. Otherwise, the paint coating would be marred by highly visible pin holes, all over the surface. Indeed, there is an even greater benefit from this precured-skin approach. It is not often recognised that the resin in co-cured sandwich panels is cured under essentially zero pressure, because the inside of the envelope bag is vented to the atmosphere and does not feel the autoclave pressure anywhere outside the core-free strips for the fasteners around the edges. Consequently, the co-cured face sheets are porous, unless a surfacing layer of adhesive film is included in the lay-up. The problem shown on the right of Figure 5 has been solved in practice by cutting back the film adhesive so that it does not flow into the corner to create a big enough blob to shrink. The problem on the left side has been solved only by precuring the detail parts and secondarily bonding them together. (The MD-11 outboard ailerons, which were of a bonded composite skin/rib design, were an improvement over the similar design for the earlier batch of R&D co-cured composite rudders for DC-10; but both were trouble-free in service – and both were lighter than metallic structures while the composite honeycomb sandwich designs of similar components were consistently heavier.) In another manifestation of this thermal-contraction-of-the-resin problem, when structures like spars and ribs are made, the angles between the web and the flanges close up during cool-down after cure and it is often necessary to compensate for this by modifying the lay-up tools. This is discussed in detail later.

All of the problems discussed in this section are easy to predict, and understand, if one thinks of composites as having discrete fibre and resin constituents, but absolutely impossible to analyse or even comprehend if one assumes that each layer, or portion, of the panel is homogeneous.

**EDGE DELAMINATIONS CAUSED BY EXCESSIVE BLOCKING OF PARALLEL PLIES**
In the absence of reliable analysis tools to predict matrix failures in composite laminates, the economic pressures to reduce costs by laying down the uncured laminate in the minimum possible number of slabs have not been restrained. And the strength and life of the cured laminates were degraded almost as fast as the cost was reduced. A series of tests was performed at Douglas Aircraft to create a warning against this malpractice many years ago, but the absence of an associated theoretical confirmation that this should have been so has allowed the concerns to be ignored by those empowered to make decisions or be held accountable for minimizing costs. The same problem arises whenever a component has been over-optimised to have too high a fraction of fibres in any one direction.

This problem is characterised in Figure 6, identifying where delaminations actually occurred.

![Diagram of delaminations](image)

**Fig. 6 Edge Delaminations, or Worse, Caused by Excessive Blocking of Parallel Plies**

Laminated blocks were constructed with the same number of effective interfaces between changes in fibre direction, regardless of the total laminate thickness. For simplicity, a quasi-isotropic laminate is shown, but the same phenomena occur with all the other laminates with different fibre fractions in each direction. According to standard homogenised lamination theory, all of these laminates should have been mathematically identical. But the testing proved that they weren’t! In every case, residual thermal stresses developed during cool-down after cure at high temperature created small stresses in the fibres, but intense stresses in the resin interfaces, which were confined to local bands in the immediate vicinity of the edges of each panel, just as happens with adhesively bonded joints. These edge stresses enforce compatibility of the strains in each layer throughout the interior of the panels – or they fail to do so whenever the effective layer thickness becomes too large. The excessive blocked ply stacks refer to tests on AS4/3501-6 carbon-epoxy cured at 180 °C (350 °F).

The problem described in Figure 6 is the reason why empirical design rules were developed to avoid this situation. Unfortunately, they haven’t always been adhered to, because they are perceived by some composite enthusiasts as an unwarranted restraint on their ability to tailor lay-
up patterns to “optimise” the structure, and by others as an unnecessary expense that could be avoided by the use of thicker plies. The optimum solutions are very different when all failure mechanisms are included in the selection process and when one of them is omitted. The design rules to overcome the deficiency in the analysis capabilities are very simple. For carbon-epoxy laminates, there should be no more than 0.010 inch (0.25 mm) of parallel fibres adjacent to a $90^\circ$ change in fibre direction, and no more than 0.020 inch (0.5 mm) adjacent to a $\pm 45^\circ$ change in fibre direction. This means that it is very difficult to create workable fibre patterns that deviate excessively from the quasi-isotropic lay-up for thick laminates. (There is more flexibility in regard to thin lightly loaded face sheets.)

**INTERNAL SPLITTING AND DELAMINATION GROWTH ASSOCIATED WITH EXCESSIVE BLOCKING OF PARALLEL PLIES**

The matrix cracking described above occurs at the edges of composite laminates. However, a different form of splitting, through each layer instead of between them, can also occur far away from the edges, in the interior of the laminates, whenever too many parallel plies are blocked together, as the result of a transverse applied load and the residual thermal stresses already in the matrix around the fibres in those excessively thick plies. These plies then split, through the thickness and parallel to the fibres, as shown in Figure 7. Then, if the adjacent stack of fibres is too thick, the cracks will not penetrate further, but change direction and become interfacial delaminations between the two excessively thick layers of fibres. This, too, is described in Figure 7. For simplicity, only a $0^\circ/90^\circ$ laminate is shown. But such internal through-the-ply cracking can occur in any thick layer in any orientation. The nature of the adjacent plies affects only what happens next, when the crack reaches the interface.

![Fig. 7 Through-the-Thickness Layer Splitting Leading to Interfacial Delaminations Caused by Excessive Blocking of Parallel Plies](image)

These phenomena, like those in Figure 6, can be analysed only using models that account for the heterogeneity of fibre-polymer composites – *AND* a failure criterion for the matrix that accounts for a combination of triaxial stresses in the matrix, and the existence of residual thermal stresses. It is simply not possible to predict these phenomena at the homogenised lamina level of analysis. But the need to do so grows at an even faster rate than the pressure increases to “reduce costs” by the mindless concept of smaller number of thicker plies in the lay-up.

**ADVERSE SECONDARY FAILURES ASSOCIATED WITH CRACK STOPPERS MADE FROM CONCENTRATED LOCAL STRIPS OF ALL-$\theta^\circ$ FIBRES**
Sometimes, the desire to create a concentrated blob of parallel fibres arises from structural considerations, rather than economic pressures. A concentrated strip of all-0° plies embedded in a more nearly quasi-isotropic laminate is known to act as a very efficient crack-stopper. However, sometimes, stopping the crack does not prevent the destruction of the component. This concept was first used, many years ago, to act as tear straps for highly pressurised, highly stressed metallic pressure vessels for space vehicles, where weight is so critical. It was found that, instead of arresting the crack in the metallic shell, it would sometimes turn the crack, which then propagated almost all the way around the circumference, as shown in Figure 8. The mechanism of this failure, (the skin expanding much more where it is not restrained by the hoop wraps), suggests that the same thing could happen with composite cylindrical pressure vessels with similar hoop-wound crack stoppers if the crack stoppers were too large or too far apart.

![Figure 8](image-url)

**Fig. 8 Unanticipated Secondary Failures Associated with Filament-Wound Crack Stoppers on Metallic Pressure Vessels**

There is a related problem to that shown in Figure 8 in the context of all-composite structures. This is the incorporation of concentrated regions of local all-0° fibres, as strips or rods, in an otherwise much softer basic laminate. This local reinforcement can be in the form of co-cured stringers or of embedded crack stoppers. Sometimes, a skin crack in the basic laminate is arrested at the edge of the so-called hard strip, but the crack turns the corner and splits the laminate right through, all the way to the end of the test panel. Even if the structure did not fail catastrophically because of this, it is usually not possible to repair the panel, particularly if it is thick, because neither the soft skin, between the hard strips, nor the strips themselves will tolerate loaded bolt holes. Indeed, the normal failure mode of bolted joints in laminates with an excessive 0° fibre fraction is not one of bearing or of net tension, but shear-out, all the way to the end, of a neat rectangular plug, regardless of how many bolt diameters away the panel end might be. These phenomena are described in Figure 9. Much the same thing happens when bolt holes are drilled in laminates with an excessive 0° fibre content. The failure mode is then the shear-out of a long narrow strip, usually but not always the width of the bolt hole, extending all the way from the fastener to the end of the panel. This was established by test as long ago as 1974, but it does not seem to have discouraged the use of excessively orthotropic fibre patterns in such members as stringer flanges and spar caps, a trap that the unwary are encouraged to fall into by the belief in optimizing the fibre pattern to maximise bending stiffness of a beam or column, for
example. The need for such caution is not always evident at the initial design stage, because there is the option of modifying the local fibre pattern by incorporating additional plies to make it closer to quasi-isotropic, in which the bolts become far more efficient in transferring load. However, that option is not available when there is a need for bolted repairs, which are the only practical possibility for thick laminates that have been damaged in service or during production.

Experience showed, many decades ago, that far more efficient crack-stopping in composite structures could be obtained by using soft high-strain fibres to replace the $0^\circ$ fibres locally. Thus, for a demonstration wing box built in the 1970s at the Douglas Aircraft Company, under contract to the U.S. Air Force, a nearly quasi-isotropic wing skin made from carbon-epoxy had the spanwise fibres replaced by fibreglass in narrow strips over each spar cap and bolted stringer location, as shown in Figure 10.
It was thus possible to locate load-carrying bolt holes in those seams. The strips also worked successfully as crack stoppers, without turning and propagating. Not only that, by keeping the bolt holes out of the basic all-carbon-epoxy laminates, except at the reinforced strips at the root and each rib-cap location, it was possible to raise the gross-section design operating strain level to 0.005 at ultimate load – a remarkable achievement then, given that the contemporary unnotched laminates failed at a strain level of only 0.008 to 0.009. The idea of the glass softening strips had originated at General Dynamics, Convair. Obviously, this design was accomplished with even more limited analysis tools than are available today, so it relied extensively on empirical wisdom. However, it had one inevitable drawback. Even a hole as small as ¼ in. (6 mm) outside any of the protected strips would decrease the ultimate panel residual strength by a factor of 2! Thus, the concept was ideal for space applications, where there is no opportunity to repair structures, but it is of limited value for transport aircraft, for example, even though it is superior in terms of repairability to the more modern alternative strips of hard and soft skin bands.

The message from all of the above examples of matrix failures and crack stoppers, which have in common the use of thick local regions of parallel fibres, is that there really is a need to be able to predict matrix failures, which are real, as distinct from predicting lamina failures that can occur only in artificially homogenised anisotropic solids, which exist only in the abstract world.

**STRUCTURALLY DEFECTIVE STIFFENER RUN-OUTS ASSOCIATED WITH OVER-SIMPLIFIED SPLICE PLATES AND INTERSECTING STIFFENERS**

On far too many occasions, stiffeners that suffer from inferior designs in which they are terminated short of the very end of skin panels or are interrupted by orthogonal stiffeners without any provision to transfer their end load directly across the interruptions in load path, delaminate from the skin. (Integrally stiffened metallic structures suffer similarly, except that the symptom then is premature fatigue cracks in the skin originating at the end of the stiffener webs.) Such a typical design is depicted in Figure 11. The failure mode is one of delamination, starting from the...
end of the stiffener and frequently progressing by tearing into the underlying skin rather than by spreading as a delamination at the interface.

**Fig. 11 Typical Example of Defective Stiffener Run-out Designs, with Hat Stiffeners**

It is never necessary to design such deficiencies into structures, as is indicated in Figure 12, which shows the demonstration panel to validate the production concept before it was modified to make it more producible. The retrofitted doublers shown in Figure 11 had to be added to a whole fleet of aircraft, because the “simplified” concept was applied to a whole class of secondary fairing panels that delaminated in service.

**Fig. 12 An Example of a Structurally Sound Stiffener Run-Out**
The expansion joints in the cross section in Figure 12 are sliding overlaps in the uncured layup to allow the rubber mandrels to expand when heated and to leave a gap during cool-down after cure as the rubber mandrel shrinks, to facilitate its extraction from the part.

Co-cured blade stiffeners terminating short of the ends of the panel represent another popular defective design concept that fails at the point of stringer runout. In lightweight panels, the failures are usually caused by transverse loads while, for primary wing and tail structures, where the problem is intensified by the much greater leads, failure is usually caused by an inability to shear the end load in the stringer into the skin. The reason why is explained in Figure 13; the stress concentration at the stiffener run-out is far higher than that associated with loaded bolt holes. The formula in Figure 13, which was derived during the CRAS R&D Contract, refers to integrally stiffened isotropic metal structures; it would be even higher for co-cured composite structures if $0^\circ$ plies had been added to the stiffeners to “enhance” their stiffnesses and strengths.

$$k_t = 1 + \frac{A_{\text{stringer}}}{t_{\text{stringer}} \times t_{\text{skin}}} \quad \text{in general, } k_t = 1 + \frac{h_{\text{stringer}}}{t_{\text{skin}}} \quad \text{for blade stiffeners}$$

**Fig. 13 Another Common Example of Defective Stiffener Run-out Designs, With Blade Stiffeners**

These failures occur because the designs are based on the false assumption that the end load would diffuse safely sideways into the skin, to be carried across the splice or interruption. It doesn’t work that way because, to do so, all of the stiffener load would need to be transferred through the single layer of brittle resin at the interface between the skin and the stiffeners. As long as there is a denial that composites are *NOT* homogeneous, this stress concentration is not exposed by customary analyses. Consequently, the need for fittings at the ends, to transfer the stiffener loads across the interruptions in load path, is recognised only by tribal knowledge of previous such failures. Sadly, this wisdom tends to disappear with each organizational restructuring.

The importance of thinking of fibre-polymer composites as a combination of *two* distinct constituents, not as of any “equivalent” artificially homogenised *single* material, is perhaps best illustrated by the following scenario. If some engineer were to propose that the aluminium stringers on the aluminium wing skin of a large wide-body transport aircraft should be
adhesively bonded together to achieve a huge reduction in part count by eliminating all of the rivets and fasteners, his suggestion would be instantly dismissed as unworkable, by both the management and most of his colleagues. Everyone “knows” that mechanical fasteners are needed for thick highly loaded structures; adhesive bonding is most efficient for thin, or laminated, structures. Yet if some other engineer were to propose that the same components be made from fibre-polymer composites – and that there was no need for all those fasteners – he would nowadays be hailed as a forward-thinking visionary, with total disregard of the fact that the brittle resin interface in the latter case between the skins and the stringers, would have less than 10 percent of the shear-transfer capability of the ductile adhesive associated with the metal-bonding suggestion. This composite concept is embraced because extremely few people understand the significance of the heterogeneity of fibre-polymer composites. Thanks to the deficient analysis tools they have been provided with, most people think of composites as homogeneous anisotropic solids. This causes the co-cured composite structures to be thought of, by most people, as equivalent to integrally stiffened machined aluminium planks. The stiffened composite panel might look like a single component when it is removed from the autoclave – and the drawing may describe it as such. But, structurally, it isn’t! It behaves like a series of discrete stiffeners, each attached to a basic skin via an ultra-thin layer of resin, which is exactly what the panel actually is. This needs to be recognised as such, and analysed as such, even if thin layers are very awkward to model for finite element analyses. It is not necessary to model each entire interfacial surface. It can safely be ignored everywhere except adjacent to any end, edge, or discontinuity. There are no potentially critical interfacial stresses anywhere in the interior of an undamaged panel. The attitude that it is not necessary to model the interface at all will persist as long as the myth of the homogeneity of fibre-polymer composites continues to be perpetuated. The weakest point in this interface is anywhere the stiffener is terminated, or interrupted, other than right at the end of a panel, where there is inevitably no load left, in either stiffener or skin. (It has already been transferred through some splice further back.) And, even at the unloaded ends, there can be delaminations between changes in fibre direction, as discussed earlier, if too many parallel plies have been blocked together.

The author has prepared an as yet unpublished paper, [1] on approximate closed-form analyses of adhesively bonded joints with geometries that do not permit exact closed-form solutions, such as tapered and scarf joints. These are based on the premise that it is permissible to ignore the relative motion between the adherends everywhere except near the very ends of the overlap. This is because there really is almost no relative motion between the adherends everywhere except at the ends of the overlap. The load is shared between the adherends in proportion to their extensional stiffnesses, $E_t$. The shear stress transmitted through the adhesive layer is then calculated by the change in end load in one or other adherend, along its length throughout the bonded overlap. What these approximate analyses yield at those ends is a concentrated load spike that does not represent reality, whenever one or the other adherend has a finite tip thickness at its end. However, if those load spikes are then applied to uniform large-area doublers of the appropriate adherend thicknesses, the exact solution for the simplified ends will coalesce with the earlier approximate analysis some distance from each end. The three analyses are then combined, to yield excellent approximations of the actual adhesive shear stresses. This process has been validated by applying it to double-lap joints with uniformly thick adherends, for which exact closed-form solutions are available. This technique covers both elastic and elastic-plastic analyses and can account for the residual thermal stresses caused by a mismatch between the adherend materials. This concept could be adapted to finite-element analyses, to eliminate the need for a fine-meshed interfacial layer everywhere except near a discontinuity.
The pressure for these unsuitably designed stiffener run-outs described earlier comes from the excessive interference in the design process from affordability initiatives, which have a history of resulting in total costs far higher than would have been incurred without that intervention. It is aggravated by the acceptance of metrics to minimise part count without proper regard for structural efficiency or actual costs. But, in the absence of analysis tools with which to demonstrate the deficiencies of these designs at the outset, even those design engineers who know not to fall into this trap seem to be powerless to prevent it. And new designers with no understanding of the heterogeneity of fibre-polymer composites study the prior art and see how frequently these inferior designs have been used in the past, apparently without raising concerns about their reliability, because it is politically incorrect to publicise them – and therefore decide to perpetuate them for their own new designs. By the time the problems have been confirmed, yet again, the opportunity to eliminate the problems, rather than merely devise a work-around, is usually lost by pressures to adhere to schedule, and by many such parts having already been manufactured and assembled. The cost of scrapping the parts, and the delay associated with a total redesign, is normally considered to be intolerable. This is why the modern literature contains few examples of alternative superior design concepts.

The author devoted an entire paper, [2] to explaining this problem, for both composite and metallic structures, and emphasised it as one of the two most important lessons he had learned throughout his 40-year career in the aircraft industry, in Ref. [3]. However, in the context of composite structures, it is obvious that these inferior designs will persist until there are analysis tools with which to quantify their deficiencies. Not even premature failures of test panels have led to the notion that there might be something fundamentally wrong with the design concept, rather than the design details. This is yet another reason why it is so important to be able to predict matrix failures accurately, early in the design process. And this process can commence only when it is acknowledged that the analysis tools available today cannot do so, even though some create the illusion that they have. More often, though, these potential failure mechanisms haven’t even been assessed, because the results of the analysis did not flag them as even existing, let alone being critical. Finite-element models are not self correcting in regard to features that had been omitted during their construction.

There are superior designs that can totally eliminate these problems, as long as the components are thin enough, and which achieve huge reductions in manufacturing cost, but only at the “expense” of an increase in part count. The best example the author can think of to confirm this is a comparison between the original co-cured tail cone on the C-17 transport aircraft and the replacement pre-cured secondarily bonded design implemented once the intolerably high costs and manufacturing difficulties of the original “low-cost” design became apparent. The original design had a solid skin stiffened by co-cured hollow-hat longitudinal stiffeners that were interrupted at each frame station to allow the mandrels to be removed from the stiffeners one segment at a time, since a full-length single mandrel would be trapped in-situ for all time. The condition in Figure 13 was thereby replicated four times at every frame stringer intersection. The replacement design relied on pre-curing simple details in the first cure cycle, that easily draped together once cured, and bonding them all together in a single second cure cycle. The manufacturing process was now so simple and reliable that there was not a single defect in any tail cone with the new design. (Some 132 such perfect components had been built by the time the author lost track of the program. Even the very first new part, which is more frequently where the problems that need fixing are exposed, was defect free and was installed on a production aircraft.) And the cost was reduced by a factor of 3! This is a great success story, documented in Ref. [4], even if it still contradicts all contemporary composites thinking.
In comparison with this outstanding success story, on another production program elsewhere, containing large complicated co-cured integral stiffeners on virtually every fuselage panel, not one large panel was made that did not need rework because of the defects in the first 58 airframes. Only the small simple panels could be made reliably. (The author has lost track of that program, too, but is unaware of any redesign, so he would suspect that there have been no improvements. The cost of two such large complex co-cured panels that needed to be scrapped nullified the entire profit made at that factory for a whole year.) The financial cost of such affordability initiatives as part-count reduction can, at times, be extremely high, because nothing can be salvaged when something goes seriously wrong. The risk is high, which is not accounted for in success-oriented planning. Sadly, this does not seem to deter further such programs.

Figures 14 and 15 compare the two C-17 tail cone designs. Figure 16 is a close-up of the bonded-beaded hollow-hat stiffeners showing how continuity of the longitudinal loads was achieved and the absence of any delamination sites at the ends of stiffeners segments – because there weren’t any. This bonded-beaded composite stiffener is a direct outgrowth of the fuselage stiffeners the author designed for the LearFan all-composite aircraft (see Ref.’s [5] and [6]). The double-thickness overlap regions shown are necessary to create slip planes in the lay-up to permit the stiffeners to be manufactured; they simultaneously provide the structural reinforcement needed to ensure that the panel compression strength is limited by only the basic design and not the stringer intersections. (Ref. [7] contains many other examples of accumulated knowledge about fibre-polymer composites, based on the author’s experiences.)

![Fig. 14 Original Co-Cured Design for C-17 Tail Cone, of High Cost Because of the Complexity of the One-Shot Fabrication](image-url)
Fig. 15 Improved Secondarily Bonded Design for C-17 Tail Cone, of Far Lower Cost but with Many More Simple Parts

Fig. 16 Close-Up of Bonded-Beaded Stiffener

One of the most intriguing premature matrix failures the author ever encountered is shown in Figure 17.
It happened during a bolted joint test conducted at McDonnell Aircraft in the 1970s, while their engineers were learning why the AV-8B composite wing needed to have a quasi-isotropic fibre pattern because the skin was bolted to the spars and ribs. In this test, additional 0° plies had been stacked together as shown in Test Coupon A, on the left, to increase the unnotched laminate strength. In earlier tests at Douglas aircraft, where the additional 0° plies had been interspersed, the shear-out plug extended all the way through the thickness, as shown on the right of Figure 18, Test Coupon B. Multiple tests were conducted, and that they showed that the shear-out failures could *not* be prevented by adding additional length to the coupons. The difference was that, in the McDonnell Aircraft tests, the 90° and ±45° plies remained unbroken throughout, except for the immediate vicinity of the bolt. All of these tests demonstrated the futility of trying to tailor the fibre pattern by adding extra 0° plies; in every such case, the gross-section strength through the bolt holes was lower than for the same bolted joint in a quasi-isotropic laminate. The author concluded that 37.5 percent was the maximum 0° fibre percentage that could be tolerated.

The matrix was just incapable of loading up any additional fibres, at least not for carbon-epoxy laminates with the low resin contents used in aerospace structures.

What all of these premature delamination problems have in common is that they will persist as long as nothing is done about the continued acceptance of the notion that fibre-polymer composites are, or behave like, homogeneous anisotropic solids.

**DELAMINATIONS IN THE CORNERS OF COMPOSITE ANGLES AND THE LIKE**

When laminated angles are made from composite layers, tape or cloth, the angle between the flanges closes up during cool-down after cure because of non-uniform shrinkage in the corner. The fibres restrain some of the resin contractions around the surface, but are unable to do so through the thickness. This is explained in Figure 18, with the contraction exaggerated for clarity. The inner and outer arcs in the corner shrink by the same proportion, but they move closer together. That is why the flanges rotate.
This problem cannot possibly be solved by fracture-mechanics analyses, since the delaminations originate away from the ends of the components.

There are no residual thermal stresses in the matrix at a stress-free temperature a little below the cure temperature, and the angle between the flanges is the same as on the tool. Then as the composite angle cools down, the angle between the flanges closes up slightly and very substantial residual tensile thermal stresses develop in the matrix. By far the greatest of these is in the longitudinal direction, parallel to any 0-deg. fibres, even though there is no associated lamina-level strain. In addition, there is a significant tensile hoop stress that develops around the fibres as the resin tries to shrink and the fibres prevent it from doing so. The residual thermal stress through the thickness of the angle is the smallest of all three components, being compressive in some places, which intensifies the tension in others. (Actually, one needs to think in terms of strains, to analyse this problem, but it is perhaps easier to visualise this case in terms of stresses.)

Now, when the flanges are pulled *apart*, significant through-the-thickness stresses develop in the matrix in the corner radius, being tensile near the inner surface and decaying and possibly becoming compressive near the outside. When the flanges are *closed up*, the opposite happens. These regions of triaxial tensile stress correspond with regions of triaxial tensile strain *in the matrix*. In other words, if one flange is bent enough relative to the other, the resin matrix will fail dilatationally in that region. This results in delaminations, which is the only way that any damage can spread; matrix cracks cannot spread in the thickness direction because any tendency to do so would be impeded by the fibres.

It is noteworthy that these delaminations *must* originate *away* from a free end of the angle because, there, the matrix must be free from longitudinal stress, so the amount of dilatation is least. In other words, fracture mechanics analyses that require, for their application, that there be a starter crack, cannot be used to analyse this class of problems.

These delaminations are real, and they can be encouraged by not compensating for the shrinkage in the loft of the lay-up tool. If that is not done, the parts are pre-loading during assembly when they are forced to fit. It may seem as if the problem is eliminated by co-curing back-to-back angles on a skin, since the upright leg remains unbent. But, actually, the inability to rotate merely prevents any relief of these residual thermal stresses and the problem is even worse.

This class of problems is almost the perfect case to highlight the deficiencies of standard composite analysis methods since the delaminations do occur, and they are impossible to even explain using the concept of homogeneous anisotropic laminae that are, by definition, free from
internal residual thermal stresses. (The phenomenon of the angle between the flanges closing up during cool-down after cure can be explained with the homogenised anisotropic plies, but not the occurrence of the delaminations.)

DELMANATIONS CAUSED BY BOLTING TOGETHER COMPOSITE PARTS THAT DON’T QUITE FIT

Delaminations can be caused when there is a mismatch between heights of adjacent parts when they are bolted together, as shown in Figure 19. The author first learned of this problem some 30 years ago when it happened on some AV-8Bs. That problem was solved, but it is still with us today elsewhere.

![Diagram](image)

**Fig. 19 Delaminations Caused By Bolting Together Composite Parts That Don’t Quite Fit**

There are three ways of responding to the occurrence of such gaps. The first is to fill the gaps with liquid shims and allow them to cure before the fasteners are tightened. But this means that the component must sit idle, with no work being done on it until the shims have set, usually for about 8 hours. (Fast-setting resins cannot be used because they have too short a working time for large parts.) The second is to fabricate individual tapered mechanical shims to fill each gap, or to sand off the high spots. This too, takes time. The third is to tighten the fasteners while bypassing these two steps, because that is common practice with metallic structures provided that the mismatches are small. (Tolerable gaps are specified in design handbooks for metallic structures.) This approach takes less time, if one does not count the subsequent repairs that are needed. Experience with composite structures has shown that the first two alternatives are far less expensive on the manufacturer, but pressure to reduce the manufacturing flow time is ever present.

The problem is the lack of any theory with which to determine how large a mismatch can be tolerated in bolted composite assemblies without shimming. This is yet another consequence of the inability to predict matrix failures in composites using any of the standard analysis tools. And it is yet another example of the appreciable costs incurred by not having such a theory.
INTERNAL FIBRE WRINKLING THAT IS NOT DETECTED
BY STANDARD ULTRASONIC INSPECTIONS
UNLESS IT IS ASSOCIATED WITH VOIDS OR POROSITY

There is another serious structural issue related to the heterogeneity of the fibre and resin constituents of composite structures that is associated more with manufacturing than with structural analysis but, this family of structural deficiencies needs to be understood, too. This is wrinkled fibres, which the author has seen caused by a number of mechanisms. It is customary to not account for any associated loss of strength in the analyses, and the condition is usually ignored, even when it is visible to the naked eye, unless it can be identified by the specified inspection tool, an ultrasonic machine to detect internal voids and porosity. The use of eyes is usually not recognised, since this is a variable process, requiring skill, and therefore unreliable, by definition, even when it is more effective.

The author’s first exposure to wrinkled fibres was a succession of co-cured designs that were difficult to make. Samples are shown in Figure 20.

![Typical Shrink Line](image)

**Fig. 20 Wrinkled Fibres Associated with Complex Co-Cured composite Structures**

Similar components were made later, defect free, by secondarily bonding together details that were easy to make “separately,” even if dozens of them might be cured in the same autoclave cycle, which is one of the ways the costs are reduced. But the most powerful cost reduction derives from not having any defects to inspect and disposition. Such parts are routinely “accepted as is,” but frequently not until after an expensive time-consuming review, because scrapping assemblies, or even sub-assemblies, is more expensive than scrapping individual details. Most of this kind of defect cannot be repaired. Certainly, it is impossible to straighten out wrinkled fibres embedded in cured resin.

The author’s next encounter with wrinkled fibres was another manufacturing problem caused by the goal of a one-shot cure of deep honeycomb sandwich panels with local blocks of high-density cure adjacent to areas where fittings were to be installed. The problem was caused by slight mismatches in the depth of adjacent blocks of core, as shown in Figure 21.
It is Necessary to Pre-Glue the Blocks of Core Together Before Machining the Core Surfaces

Composite Skin

Wrinkled Fibres in Region of Zero Pressure

Caul Plate

Autoclave Pressure

High-Density Core Block, as Around Fittings

Foaming Adhesive Core Splice

Low-Density Core

Fig. 21 Wrinkled Fibres Associated with Mismatches in the Height of Adjacent Blocks of Core

When the outer surface was defined by a stiff caul plate, there would be a band around any block of core that was deeper than adjacent blocks in which there was no pressure on the fibres to hold them down during the cure. If the mismatch was extreme, the laminate would be porous there. But usually, it was just an issue of fibre wrinkling that was visible to the naked eye, but which could not be detected by ultrasonic inspection. This condition is referred to as marbling. The solution was to pre-bond the fittings and blocks, without the skins, and to machine the core to remove mismatches. Doing so produced defect-free components from then on – which were less expensive, even though manufacturing now required a two-step process, because there were no defects and none of the delays, inspections, and repairs associated with them.

The problem with this kind of defect, the wrinkled fibres, is that one knows that they are structurally inferior to an equivalent laminate without the wrinkles, but there are no analysis tools with which to characterise the loss of strength caused by the wrinkles – particularly if one is using a model that requires that the composite parts be homogeneous, even though they aren’t! Worse, unless the problem is so bad as to cause voids or porosity, the standard ultrasonic inspections will not even flag this as a defect. In this case, the appropriate response is to improve the manufacturing process. Being able to predict the inferior strength of laminates with wrinkled fibres once the parts had been cured would not be of any value.

CONCLUDING REMARKS

The author hopes that the practical examples described here confirm the case presented in Part 1 that it is necessary to be able to predict matrix failures in the so-called composite materials reliably – and, further, that there is no possibility of doing so with theories that have been oversimplified by the never-verified assumption that it was permissible to homogenise the discrete fibre and resin constituents into an “equivalent” anisotropic solid. It is then not possible to account for the residual thermal stresses in the resin matrix, because there are no terms in the theory to allow the existence of internal stresses in the absence of any external mechanical forces. This is normally compounded by the further unjustified assumption of interactions between unrelated failure mechanisms in the composite laminates, and the need to replace the
measured transverse ply properties with others that cannot be measured, but which are selected arbitrarily to provide more acceptable predicted laminate strengths.

In addition to reinforcing the case that there really is a need for scientifically valid composite material failure models that can actually predict matrix failures on the basis of measured properties, under all fibre volume fractions and operating temperatures, the illustrative examples are intended to warn the composites users where matrix failures are likely to occur. They are also meant to illustrate the features, such as residual thermal stresses, that must be included in all future composite failure theories, if there are any more possible after SIFT, which appears to be definitive, if such new theories are to have any relevance to matrix failures. The illustrative examples also expose the unanticipated negative consequences of many of the fashionable composites design techniques that have proliferated in the absence of any theories with which to warn of their downsides – before they are exposed far too late by structural testing.

Perhaps the most important message from these examples in Part 2 is that it is absolutely impossible to even comprehend any of them without first abandoning the traditional concept of homogeneous anisotropic materials and replacing it with acknowledgement that fibre-polymer composites really do contain discrete fibre and resin matrix constituents that don’t necessarily both fail simultaneously. Not only that, local failures in either constituent do not change the thermo-mechanical properties of the undamaged remainder of each constituent. If the further use of the bogus theories is banned by at least the victims who suffered the most from them in the past, the next generation of composite structures might not contain so many unanticipated premature matrix failures.

REFERENCES


