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Innovative Design of Composite Structures:
Use of Curvilinear Fiber Format to Improve
Structural Efficiency

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M.W. Hyer
R.F. Charette

Department of Mechanical Engineering
University of Maryland
College Park, MD 20742

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ABSTRACT

To increase the effectiveness and efficiency of fiber-reinforced materials, the idea of using fibers in a curvilinear rather than the traditional straightline format is explored. The capacity of a laminated square plate with a central circular hole loaded in tension is investigated. The orientation of the fibers in some or all of the layers is allowed to vary from point to point in the laminate. The orientation of the fibers is chosen so that the fibers in a particular layer are aligned with the principal stress directions in that layer. Finite-elements and an iteration scheme are used to find the fiber orientation. A noninteracting maximum strain criterion is used to predict load capacity. The load capacities of several plates with different curvilinear fibers format are compared with the capacities of a more conventional straightline format designs. It is found that the most practical curvilinear design sandwiches a group of fibers in a curvilinear format between a pair of $\pm 45°$ layers. This design has a 60% greater load capacity than a conventional quasi-isotropic design with the same number of layers. The ± 45 layers are necessary to prevent matrix cracking in the curvilinear layers due to stresses perpendicular to the fibers in those layers. The results of the study indicate greater efficiencies can be achieved with composite structures than are now being realized.

INTRODUCTION

Advanced fiber-reinforced composite materials were introduced roughly two decades ago as a high performance structural material with a vast potential for improving and perhaps even redefining structural efficiency and performance. In the decade that followed their introduction, the majority of the attention given to advanced composite materials focused on learning to produce the material at a reasonable cost and learning to fabricate simple structural shapes from the materials. Attention was also directed to analysing and predicting the response of these simple structural shapes. At that time the rapid evolution of matrix and fiber materials was taking place and as a result, much attention was given to the material science aspects of advanced composite materials.

In the next decade attention focused on understanding the complex and often unexpected failure mechanisms of these advanced materials. In addition, more complex structural shapes were fabricated. Because there are so many variables available with composite materials that influence structural response, many of the components that were fabricated were designed to respond somewhat like their metallic counterpart, namely, isotropic material behavior. This is opposed to designing components which exhibit many of the elastic couplings that are available when using composite materials. This approach was acceptable at that time since fabrication itself was a major focus, and since even with metal-like construction the lower density of composite materials afforded a weight savings. And it appeared that the use of composite materials could offer a savings in the manufacturing area. These two issues were enough to stimulate interest in advanced composite materials, even if the material was being used to respond like a metal.

Since the introduction and initial development, advanced composite materials have been used successfully in many military aircraft and both military and civilian spacecraft. Currently fiber-reinforced materials are being used in business aircraft and to some extent, commercial aircraft. Despite the increased usage, there is a strong feeling among some researchers that the potential advantages of advanced fiber-reinforced composite materials have not been fully exploited, or even explored. The development and use of advanced composites in structures has followed a rather slow evolutionary course rather than a quantum increment approach. As a result, some of the advantages of composite materials have been eroded by some of their detrimental characteristics. The result is, in some instances, a marginal gain, compared to a metallic, when composite materials are used. For example, a quasi-isotropic laminate has stiffness characteristics much like aluminum and has similiar inplane load capacity. However, unlike aluminum, a quasi-isotropic laminate can delaminate. Aluminum has no such problem. A quasi-isotropic laminate is susceptible to environmental degradation. Aluminum has less serious problems with this. Moreover, the manufacture and repair of composite materials is less fully understood than the manufacture and repair of aluminum. When more weight is added to a composite structure to compensate for these problems, the gains by using composite materials in a conservative fashion begin to decrease. When all things are considered, a composite material may not be the final choice. This is particularly true when one considers the increases in performance of the new lithium aluminum alloys. In the final analysis, with current philosophies regarding the use of advanced composite materials, the gains being discussed are in the 15-20% range, the evolutionary level, as opposed to gains of a factor of 2 or 3, the revolutionary level. The only way to overcome this problem is to begin to abandon the overly conservative utilization of a material which seems to have considerably more potential. Some of the design philosophies and traditions developed during the last two decades must be reexamined and perhaps even ignored altogether. The work reported on here represents a departure from traditional usage of composite materials and attempts to explore the idea of using composite materials more effectively and perhaps realize step increases in structural performance. Specifically, this study examines the issue of using fiber reinforcing in such a manner that the direction of the fibers, or at least some of the fibers, is a function of spatial position. Herein this is referred to as a curvilinear fiber format. In fact, if one has at their disposal, strong, stiff fibers for reinforcing a structure, it is not altogether clear that aligning the fibers parallel to each other and in straight lines is the best way to utilize the fiber. It is true that in the past this is the format in which fibers have been supplied. The fibers have been impregnated with resin, aligned parallel with each other, and rolled on to a spool. The user makes a structural component by stacking together multiple layers of straight and parallel fibers, the fibers in each layer being in a straight line. Yet it is possible that the component contains a geometric discontinuity, such as a hole, that interupts fiber continutity in all of the layers and causes a concentration and realignment of the stresses. If it could be done, it would seem that not breaking fiber continuity, and somehow using the fibers to advantage near the geometric discontinuity would seem the efficient way to use strong stiff fibers to advantage. In particular, it would seem that the fibers should "flow" continuously around the discontinuity and be oriented in such a way as to transmit the load efficiently around the discontinuity. To be sure, there are many issues that must be studied with this idea, the most important being fabrication. However, the availability of raw fiber, the increased power and flexibility of robotics, and new matrix development do provide promise for fabricating components on a fiber-by-fiber basis with something other than a straightline format. These promising fabrication techniques aside, researchers would be remiss if only straightline fiber format were considered.

The particular problem studied here is a prime candidate problem for deviating from the straightline fiber format. The particular problem is a plate loaded in its plane and containing a central circular hole. The problem has been studied hundreds of times by many researchers dealing with isotropic and composite materials. It has been studied to some extent in the context of curvilinear reinforcement[1,2]. It has been studied a number of times because it is an important problem, in general. Here it is studied specifically because aircraft structures contain many holes for access and fabrication, in addition to numerous windows if commercial transports are being considered. These are all discontinuities in otherwise continuous structures and they can lead to inefficiencies in the use of material. In this report the design of a plate to resist inplane tensile loads is discussed. Inplane compressive buckling loads are clearly of interest and they are to be studied in a subsequent phase of the work. The sections that follow discuss the problem specifics and some of the issues related to using a curvilinear fiber format. Numerical results which illustrate the increased efficiency, relative to conventional straightline design philosophies, of a curvilinear fiber format are presented.

PROBLEM DESCRIPTION

Figure 1 shows a square plate with a central circular hole. The plate width, and length, W, is three times the hole diameter, D, i.e., W/D=3. Shown is a uniform tensile load applied at opposite ends of the plate. The basic issue is as follows: Given that a graphite-epoxy material with a fiber volume fraction of roughly 65% is available, how can the material be used most effectively to construct a plate with the dimensions and loading of fig. 1? The term "most effectively" implies that the load transmitted is the maximum, or that the weight of the plate is a minimum. That the words 'maximum' and 'minimum' are being used is unfortunate because it implies something akin to optimal design is being studied. Optimum design is not the issue being discussed. The issue being discussed is to use fundamental knowledge regarding the response of fiber-reinforced composite materials, coupled with limited analysis, to design the plate to carry more load than conventional straightline fiber formats allow. The designs focus specifically on layered plates with fibers which vary direction within a layer or group of layers.

DESIGN PHILOSOPHY

The basic philosophy used to design the plate is to assume that the fibers should, in some sense, be aligned with the principal stress directions in the plate. Strictly speaking, principal stress directions are meaningless when discussing fiber-reinforced materials. More meaningful are the principal material directions. However, here, principal material directions, in conjunction with principal stress directions will be utilized. Specifically, in certain layers the principal material directions of the layer and the principal stress directions of the layer will be aligned.

METHOD OF ANALYSIS

All numerical calculations discussed here are based on a finite-element analysis of the loaded plate. The element used is a standard 8-node isoparametric element. There are 9 Gauss integration points per element. Figure 2 shows the mesh used in the finite-element analysis. The mesh assumes quarter symmetry in the response, a valid assumption considering only inplane response is being studied. The quarter model uses 120 elements. Fiber orientation is defined relative to the +x axis. A positive angle for fiber orientation corresponds to counterclockwise rotation from the +x axis. Here the inplane strength of the composite material is evaluated using a noninteracting maximum strain criterion. Lamina strains in the fiber direction, perpendicular to the fibers, and in shear are used as indicators of failure. Other failure criteria could be used but the important point is to use the same criterion on all designs. This provides a meaningful comparison of designs.

The material properties used throughout the study are chosen to closely represent those for AS4/3501. The elastic properties are given by:

$$E_1 = 19.9X10^6 \text{ psi} \quad E_2 = 1.28X10^6 \quad G_{12} = 1.03X10^6 \quad \nu_{12} = 0.298$$

$$\text{ply thickness} = 0.005 \text{ in.}$$

The failure strains used are as follows:

tensile failure in fiber direction, $\varepsilon_{1T} : 10.5X10^{-3}$,
compression failure in fiber direction, $\varepsilon_{1C} : 10.5X10^{-3}$,
tensile failure perpendicular to the fiber, $\varepsilon_{2T} : 5.8X10^{-3}$,
compression failure in perpendicular to the fiber, $\varepsilon_{2C} : 23.0X10^{-3}$,
shear failure in plane of lamina, $\gamma_S : 13.1X10^{-3}$.

The strains used to examine failure are determined by using the stresses at a particular Gauss point in conjunction with the stress-strain behavior of the material. The same Gauss point is used for the strain calculations for each element.

CONVENTIONAL DESIGNS

Conventional designs of structural components using fiber-reinforced composite materials orient the fibers in the various layers so as to result in a quasi-isotropic or mildly orthotropic configuration. The quasi-isotropic design is used here as a basis of comparison. The strength of other designs are compared with the strength of the quasi-isotropic design. The particular quasi-isotropic configuration is 16 layers with a stacking sequence of $(\pm 45/0/90)_{2s}$.

With the above material in the above quasi-isotropic configuration, the finite-element analysis and the maximum strain failure criterion predict failure at the net-section hole edge in the 0° layers. Failure is predicted to occur at a load of 1472 lb./in. of plate width. Henceforth everything will be normalized by this load and so the load to cause failure of a quasi-isotropic laminate is unity. The analysis also indicates that the matrix in the 90° layers fails at the net-section hole edge at a load level of 0.54. This failure is not considered important because the fibers in all the layers are still intact at this load level.

Another fairly conventional and current design philosophy incorporates two 0° layers rather than one 0° and one 90° layer. This results in a layup of $(\pm 45/0_2)_{2s}$. Such a laminate exhibits mildly orthotropic behavior. Clearly with more fibers in the load carrying direction, the load capacity should be greater than the quasi-isotropic design. An analysis of the mildly orthotropic case indicates that the fibers at the net section fail in the 0° layers at a load level of 1.17. The matrix in the ±45° layers fails in tension perpendicular to the fibers and shear at a load levels close to 0.93. Again, with fibers being intact at this lower load level, the matrix cracking is not considered detrimental to the integrity of the plate.

A somewhat more unusual design, but still utilizing fibers in a straight line, is a $(\pm 45/0_6)_s$ laminate. This laminate is highly orthotropic and is included in this study so that the limitations of fiber strength in a straightline format are fully explored. The finite-element analysis of this laminate indicates net-section fiber failure in the 0° layers at a load of 1.24. This represents a 24% increase over the quasi-isotropic design. The analysis shows that the +45° layer fails in matrix tension at a load level of 0.69. As has been done previously, this matrix failure is dismissed as being unimportant. Table 1 summarizes the findings of the failure analysis of these three conventional laminates.

TABLE 1

Failure Loads of Conventional Designs

Design	Failure Load	Failure Mode
$(\pm 45/0/90)_{2s}$	1.00	fiber failure in 0° layer at net-section hole edge
$(\pm 45/0_2)_{2s}$	1.17	fiber failure in 0° layer at net-section hole edge
$(\pm 45/0_6)_s$	1.24	fiber failure in 0° layer at net-section hole edge
$(0_8)_s$	0.612	shear failure in matrix near hole edge, away from net section

To provide an additional comparison, one other straightline format laminate is studied. The laminate is simply a $(0_8)_s$ laminate, all 16 layers having their fibers aligned with the load. The design is unusual and impractical but will be used shortly for comparison with the curvilinear format. The analysis indicates that the matrix fails in shear near the hole edge, away from the net-section, at a load of 0.61. At the location of failure the stresses are changing rapidly and so inevitably shear stresses are high. Since the unidirectional stresses are inherently weak in shear, failure occurs. For this design, matrix failure is important and considered to be laminate failure. Without the matrix intact, the unidirectional material would simply disintegrate. Table 1 also includes the results for the $(0_8)_s$ laminate.

CURVILINEAR DESIGN

The first idea that comes to mind with the notion of using a curvilinear fiber format is to orient the fibers in the principal stress directions. It must be kept in mind that principal stress directions depend on material properties. What may be thought of as principal stress directions for an isotropic material, such as aluminum, will change when fibers are introduced into the material. In addition, the idea of principal stress directions for a laminate must be applied carefully. Each layer has principal stress directions and these directions vary from layer to layer. Here an iterative approach is used to find the fiber directions within a group of layers such that the fibers are everywhere aligned with the principal stress directions in these layers. The first iteration assumes that the fibers are everywhere aligned with the principal stress directions for an isotropic plate. Subsequent iterations are used to align the fibers with the actual principal stress directions.

With the finite-element discretization used here, several approximations are inherent in the analysis. First, it is assumed that within an element the fiber direction is constant. Second, the calculations for principal stress directions and fiber directions key on the stresses computed at just one Gauss point in the element. There are nine points to choose from and the choice is somewhat arbitrary. Initially the point at the centroid of the element was used. To obtain stresses closer to the hole edge, calculations using the centroidal Gauss point were redone using a Gauss point in each element which resulted in using stresses closer to the hole edge. Qualitatively the results using the two Gauss points were the same. Quantitatively the the loads to cause failure were different. However, since all loads were normalized by the load to cause failure of a quasi-isotropic laminate, the quantitative differences were eliminated. Finally, there is a problem with the principal stress calculations at the so-called isotropic point at the hole edge. The isotropic point is as follows: The circumferential stress around the hole changes from tension at the net section to compression on the horizontal centerline of the plate. At some point around the circumference, the circumferential stress is zero. Since the hole edge is traction free, all stresses are zero at this point. Thus, all directions are principal stress directions at the isotropic point. Obviously principal stress directions are meaningless at this point and so the fibers direction at this point is chosen to be consistent with the directions of neighboring elements.

With the fibers in each element aligned with the principal stress directions of the element from the isotropic analysis, the stresses are recomputed and another set of element principal stress directions determined. The fibers are aligned with these new directions and the stress analysis repeated. A third set of element principal stress directions is computed and the process of fiber realignment is repeated until the principal stress directions and the fiber directions are aligned to within a given tolerance. Specifically, the solution is considered converged when the quantity

$$\frac{\theta - \phi}{\theta}$$

is less than 0.01 for each element. In the above θ denotes the principal stress direction and ϕ denotes the fiber orientation. In practice, convergence was realized in 4 or 5 interations. For the laminates studied here, the result of the iteration process was a principal stress direction for each element that was not significantly different than the direction from the isotropic analyis.

The first curvilinear design discussed aligns all 16 layers with the principal stress direction: an all-curvilinear design. This laminate is denoted $(C_8)_s$, the C standing for curvilinear. This laminate is the curvilinear counterpart to the $(0_8)_s$ laminate previously discussed. Since the $(0_8)_s$ laminate failed due to shearing of the matrix, aligning the fibers with the principal stress directions would eliminate that problem. The finite-element analysis of the $(C_8)_s$ laminate confirms this. The analysis predicts that the fibers fail in tension at the net-section hole edge at a load, relative to the quasi-isotropic case, of 2.01. However, the matrix is predicted to fail in tension at the net section, but away from the hole edge, at a load of 0.9. Since there are no fibers in another layer to supress the matrix failure, this matrix failure constitutes failure of the laminate. Indeed, with the matrix cracking, the laminate will desintegrate, much like the $(0_8)_s$ above, with many cracks in the matrix following the fiber direction. Using this as a failure criterion, the $(C_8)_s$ design is weaker than the quasi-isotropic design and the 2.01 load level is considered a hypothetical ideal. Table 2 summarizes these findings for the all-curvilinear design, as well as summarizing the results discussed next.

TABLE 2

Failure Loads of Curvilinear Designs

Design	Failure Load	Failure Mode
$(C_8)_s$	0.9	tension perpendicular to fiber direction at net-section, away from hole edge
$(O/C_7)_s$	1.96	fiber failure in curvilinear layers at net-section hole edge
$(\pm 45/C_6)_s$	1.61	fiber failure in curvilinear layers at net-section hole edge
$(O_2/C_6)_s$	1.87	fiber failure in curvilinear layers at net-section hole edge

The next logical step in the curvilinear design is to eliminate the matrix tension failure in the $(C_8)_s$ laminate. If this can be done, then the load capacity of the plate would be increased by a significant factor relative to the quasi-isotropic plate. The matrix tension cracking in the $(C_8)_s$ is the result of a Poisson effect in the plate. The high strain in the x-direction at the net-section hole edge causes a large contraction strain in the y-direction at the hole edge. Away from the hole edge the net-section strain in the x-direction is not as large and as a result, the Poisson contraction in the y-direction is not as large. With a large y-direction Poisson contraction at the hole edge and a smaller y-direction Poisson contraction away from the hole edge, the material is acutally subjected to a tensile stress in the y-direction. This tension is enough to crack the matrix. If a small amount of reinforcment is added perpendicuar to the fibers in this region, the matrix tension cracking would be supressed there. The matrix cracking would subsequently occur at some other location at a load below that required to fail the fibers at the net section at the 2.01 load level. A possible solution to this idea of "chasing" the matrix tension cracking around the laminate is to add one or more layers of fibers that are everywhere perpendicular to the original 16 curvilinear layers. Thus the laminate would be an orthogonal grid of fibers. This design is pursued next.

To keep the laminate to 16 layers, so the weight and thickness are identical to the other laminates in the study, 14 layers are chosen to be curvilinear in the primary load direction and 2 are chosen to be orthogonal to these layers. The notation adopted is $(O/C_7)_s$, the O denoting layers orthogonal to the curvilinear layers. Though it is felt this change in the laminate would not significantly alter the principal stress directions relative to the $(C_8)_s$ laminate, the iteration procedure was repeated, starting with the fiber directions for the $(C_8)_s$ design. As expected, after the iteration produced a converged solution, and the fiber directions of the $(O/C_7)_s$ and the $(C_8)_s$ were practially identical.

The addition of the orthogonal layers more than supresses the matrix tension cracking. The $(O/C_7)_s$ laminate is predicted to fail at a load of 1.96, the fibers at the net-section hole edge limiting the load. Matrix tension cracking occurs at a much higher load. The lower value of load for the $(O/C_7)_s$ compared to the 2.01 ideal load of the $(C_8)_s$ laminate is the result of converting two of the 16 layers in the $(C_8)_s$ laminate to O layers, thus eliminating two of the load-bearing curvilinear fiber layers.

Though the $(O/C_7)_s$ laminate accomplishes the goal of preventing matrix failure, it suffers from two distinct problems. First, it is very susceptible to a shear failure. Any slight misalignment of the tensile loading, or any amount of shear introduced at the plate boundaries, would most likely crack the matrix. The likelihood of having a pure tensile loading, as has been assumed, is small and so shear is an issue. Second, the manufacturing of a laminate with an orthogonal grid of layers could be a problem. It is probably safe to say that any manufacturing technique would result in less than perfect placement of the fibers. The fibers would be oriented in the desired direction to within a specified tolerance. Because of this slight misalignment of fibers, there could be unwanted tensile or shear stresses in the matrix. This would lead to matrix failure and negate any gains made by using curvilinear fibers. To counter any unwanted loading or any misalignment of the fibers, a pair of $\pm45°$ layers are used in the laminate instead of using an orthogonal layer arrangement. The result is a $(\pm45/C_6)_s$ laminate. Adding the $\pm45°$ layers to a laminate would be straightforward from a manufacturing viewpoint. Furthermore, the off-axis layers tend to serve as a sandwich to hold the curvilinear layers together, a desirable feature. With the off-axis layers in the laminate, the direction of the curvilinear layers would be different than the direction of the curvilinear layers with the orthogonal layers. Furthermore, the fibers in the curvilinear layers should be aligned with the principal stress directions of the curvilinear layer, not the principal stress directions of the laminate. Therefore the iteration procedure is used to find a solution.

The iteration procedure applied to the $(\pm45/C_6)_s$ laminate resulted in a curvilinear fiber format that was not appreciably different than the curvilinear format for the $(C_8)_s$ and $(O/C_7)_s$ designs. The analysis indicates that the $(\pm45/C_6)_s$ laminate sustains a load level of 1.61. Failure is due to tension in the fiber direction in the curvilinear layers at the net-section hole edge. There is matrix shear failure in the $\pm45°$ layers near the hole edge at a load level of 1.36 but this is not considered detrimental to the integrity of the laminate.

Figure 3 illustrates the directions of the curvilinear fibers in the $(\pm45/C_6)_s$ design. The figure is presented to illustrate several important points. First, the figure illustrates the assumption in the analysis that the fiber direction within each element is constant. Even with this restriction, the fiber trajectories are smooth. Second, the figure illustrates that the fiber trajectories involve fairly gentle curvatures. For a structure with a hole 8-12 in. in diameter, radical changes in fiber direction with location do not occur. It is therefore conceivable that such a laminate can be manufactured.

The isotropic point mentioned above occurs near point A in the fig. 3. As stated earlier, at this point all directions are principal stress directions and the solution scheme has difficulty converging to a fiber direction. Also, from point A to point B (the hole edge on the centerline) the circumferential stress is compressive and the radial stress is tension but is small. The iteration scheme choses fiber directions in the direction of the compressive circumferential stresses. As a result of these two characteristics of the stresses in the region near arc AB, the fiber directions choosen by the iteration scheme were not consistent with the fiber directions in the surrounding elements. Therefore, the fiber directions in this area were adjusted by hand so they would be consistent with fiber directions in the surrounding area. The magnitude of the stresses in this region are small and so the fiber direction was not critical. In addition, the ±45 layers help react the stresses in this region.

Another important point is to be made regarding the fiber directions illustrated in fig. 3. To impliment the idea of a curvilinear fiber format, the fiber directions computed in the analysis must be translated into trajectories for actual placement of the fibers in the manufacturing of plates. This step has not been done to date but such a step is planned. Figure 3 shows several hand-faired fiber trajectories to illustrate the importance of this step. It is felt that once the trajectories are determined, the fiber direction at any particular point may be slightly different than the direction called for by the iteration analysis. This will be due to the fact that the fiber directions have been assumed constant within an element when in fact the actual trajectories will be continuously changing direction. However, it is felt that the 4 off-axis layers will be enough to react any unwanted stresses that result from perturbations in fiber trajectories.

Finally, one other curvilinear design is explored. The $(O/C_7)_s$ laminate has 14 load-bearing layers and two layers to supress matrix failures in the load-bearing layers. The $(\pm 45/C_6)_s$ laminate has only 12 load-bearing layers. Though the ±45° layers may carry some of the load, they are there mainly to supress matrix cracking in the curvilinear layers and react any unwanted shear loading. The $(O/C_7)_s$ laminate is predicted to carry more load than the $(\pm 45/C_6)_s$. Is this due to the larger number of primary load-bearing layers? To answer this question, a $(O_2/C_6)_s$ laminate was studied. This laminate also has only 12 load-bearing layers, the 4 other layers being used to supress matrix failure. Numerical results indicate that the $(O_2/C_6)_s$ laminate fails at a load of 1.87, 16% more than the $(\pm 45/C_6)_s$ laminate. This is somewhat suprising. It was felt before analysing the $(O_2/C_6)_s$ laminate that it would sustain less load than the $(\pm 45/C_6)_s$ laminate. It is not clear why there is this difference. Another comparison leads to a similiar interesting question. Near the net section, the $(\pm 45/0_6)_s$ and $(\pm 45/C_6)_s$ laminates have similiar construction. Each laminate has 12 layers with fibers close to being parallel to the x-axis and a pair of (±45) layers. Yet the $(\pm 45/C_6)_s$ design sustains 30% more load than the $(\pm 45/0_6)_s$ design. It is not clear why this should be the case either. Currently these questions are being explored.

FUTURE ACTIVITIES

The next major activity in this study is to determine the influence of the curvilinear design on compressive loading, specifically, buckling. Iteration will not be involved. Essentially the goal is to predict the buckling loads of the 8 laminates in Tables 1 and 2 and determine specifically if the curvilinear designs lead to better buckling performance. If both the tensile and compressive behavior of curvilinear designs appear to be improved, then serious consideration will be given to fabrication and testing of such plates.

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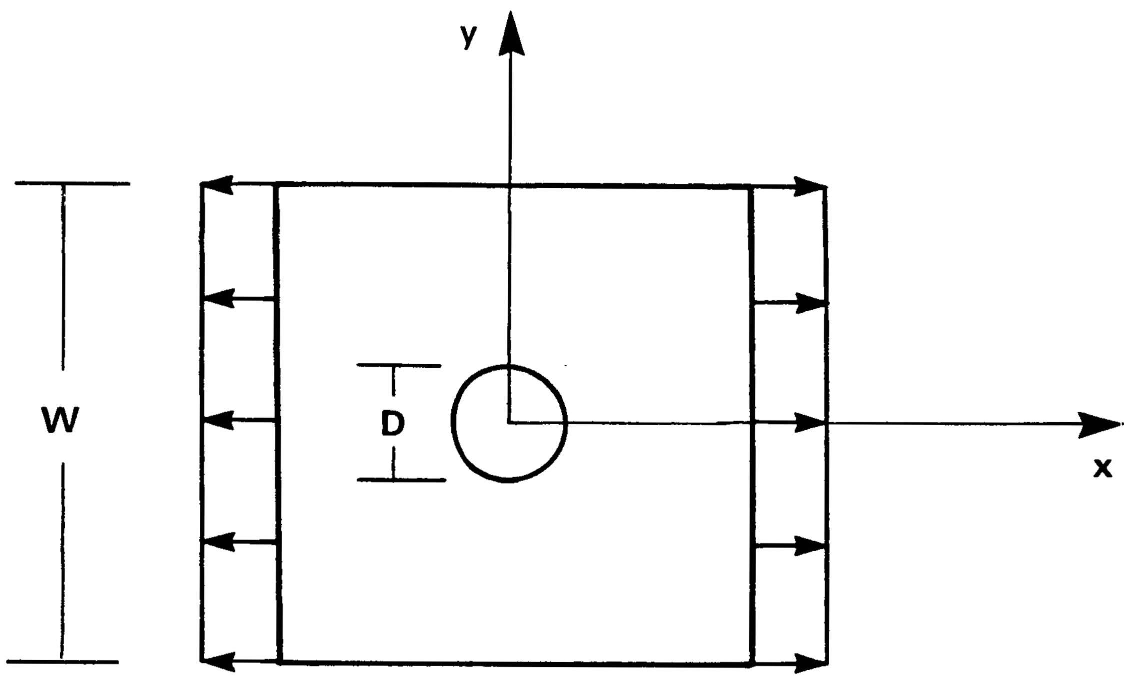

Fig. 1 Plate Dimensions and Loading.

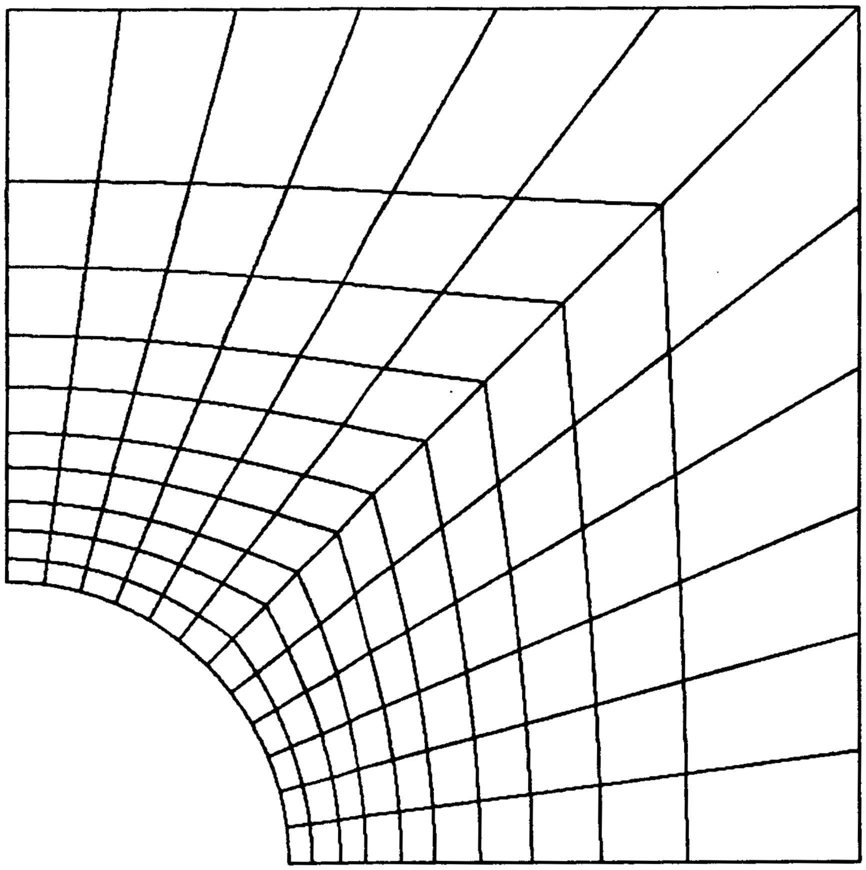

Fig. 2 Finite Element Grid.

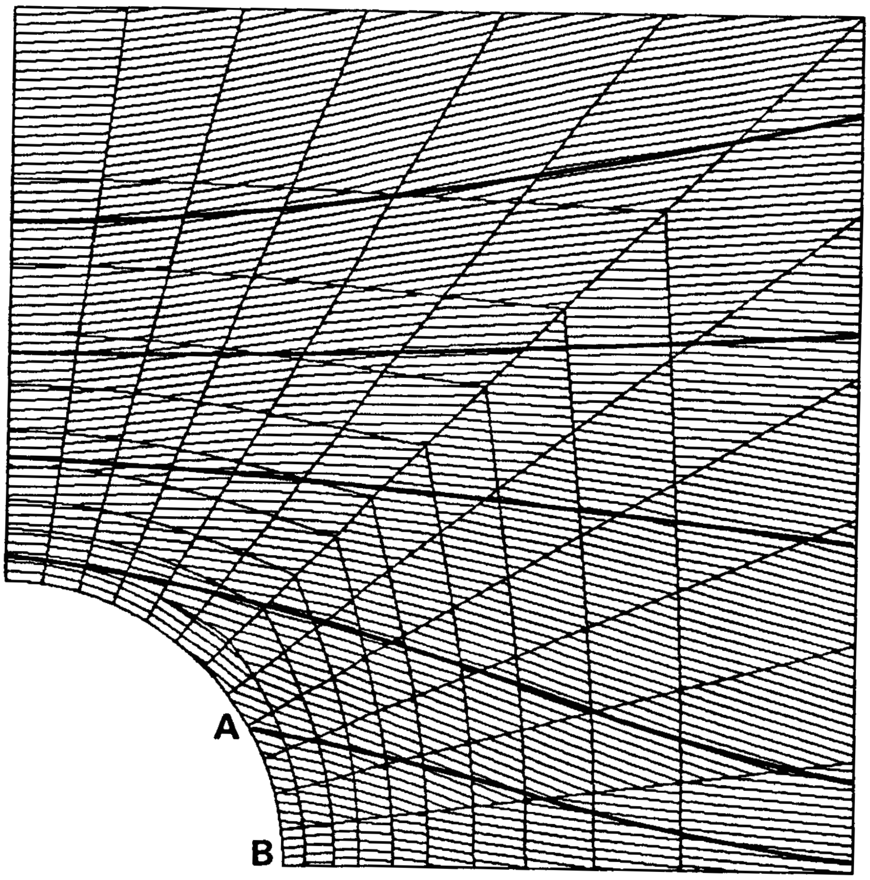

Fig. 3 Directions of Curvilinear Fibers.