Delamination-Based Approach Toward Fracture Control of Composite Spacecraft Structures

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Delamination fracture is a common failure mode in composite structures, particularly those used in spacecraft structures. Such structures are generally designed for high stiffness and dimensional stability. To meet these requirements, spacecraft structures have a high degree of anisotropy. Such construction results in laminates that have low transverse strength and are susceptible to delamination. The INTELSAT V and INTELSAT VI communication satellites, for example, use I-beam stiffeners of high-modulus graphite fiber to stiffen thin composite sandwich shells. These structures are usually well analyzed for in-plane loads arising from the launch vibration environment. But small out-of-plane loads can cause delamination. Such loads can arise from second-order geometric effects, handling, or supporting minor attachment brackets for thermal control. Incorporating fracture criteria in defining the minimum margins of safety of the entire structure will prevent such unanticipated failures. In this paper, the elements of a fracture control plan are proposed in which delamination fracture toughness is used as a material strength parameter. Currently, no such plans exist for composite spacecraft structures.

Nomenclature

a	= half-crack length
a_c	= critical half-crack length
C	= specimen compliance
C_1	= empirical constant
E	= Young's modulus
E_1	= longitudinal modulus
E_2	= transverse modulus
G_{Ic}	= opening mode (mode I) critical crack energy release rate
G_{12}	= in-plane shear modulus
M_c	= critical bending moment for catastrophic crack propagation
N	= number of cycles
n	= empirical constant
P_c	= critical load at which crack propagation begins
w	= width of specimen
δ	= crack-opening displacement
v_{12}	= in-plane Poisson's ratio
θ	= change in angle of the constant stiffness beam flange

Introduction

THE fracture-control plan for the U.S. Space Transportation System (STS), or the Space Shuttle, and its payload specifies requirements for analysis and inspection techniques for ensuring that critical structures have adequate margins of safety. This procedure is based on assuming the pre-existence of flaws or cracks of sizes smaller than those detectible with a high degree of confidence. Details of these Shuttle payload requirements are specified in Ref. 1. In this reference, fracture-control requirements are applied to metals. Composite materials are specifically exempt from fracture-control requirements.

With the growing use of composite structures, however, it may be advisable to address the issue of a fracture-control plan for composite structures. Composites now constitute the primary structures of payloads, such as communications satellites, space probes, and instrumentation packages, such as the space telescope. Failure in a primary structure can constitute a severe safety hazard, particularly if such a failure interferes with safe deployment of the payload from the Shuttle bay, or if failure results in detachment of a structure that penetrates the envelope of the payload.

Currently, composite structures are qualified for launch through typical qualification and acceptance test procedures that all structures (i.e., metallic structures), subsystems, and systems must undergo. As mentioned earlier, fracture-control requirements, which are required for critical metallic structures, are not applied to composite structures.

A fracture-control plan for composite structures has serveral benefits:

1) It would provide for a fail-safe approach to the design of a primary spacecraft structure.

2) It integrates available inspection techniques with proof testing to define the minimum flaw that can exist in such a structure without failure.

3) It can be used to define the useful life for reusable structures such as the cradle used to support payloads within the Shuttle bay.

4) It can be used to predict the sequence and modes of failure of a primary structure, making it useful in failue mode effects analysis.

A fracture-control plan must contain a discussion of suitable nondestructive evaluation (NDE) techniques, since the criterion for flaw detection depends on the type of NDE technique used. Most techniques are efficient at detecting delamination-type flaws in composites, because such flaws are oriented parallel to the lamination plane that is obviously known, and because thin laminates are typically used in spacecraft construction. Examples of appropriate techniques are ultrasonic attenuation measurements, through-transmission ultrasonic C-scan, and pulse-echo ultrasonic methods.[2-4]

This argument leads to a recommendation for the application of fracture mechanics criteria for composite spacecraft structures based on delamination fracture toughness and appropriate NDE techniques. This paper recommends the incorporation of a fracture-control plan based on delamination fracture criteria and appropriate inspection criteria. This is the first step toward ensuring the design and construction of safe, damage-tolerant composite structures for space applications.

Fracture in Composite Materials

Engineering structural laminates under in-plane or bending loading are designed using conventional laminated plate

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theory coupled with a suitable failure criterion such as maximum strain or the quadratic failure criterion. The analysis of such laminates is easily done, and the margins of safety can be determined with reasonable confidence. However, failure due to the propagation of a pre-exisiting flaw in such laminates is difficult to predict because the anisotropy of the material, local stress state effects caused by material heterogeneity, and the variety of fracture modes that exist in composite materials.

General fracture processes in composite materials are complex. Damage can develop in the form of fiber breakage, fiber debonding, delamination, matrix microcracking, and other microfailure modes.[5] It is for this reason, and because of the difficulty in analyzing and detecting these modes, that a fracture-control plan for composites has not been developed. The situation, however, is not hopeless, since the majority of failures in engineering laminates (defined here as those that contain at least three fiber orientations and that are orthotropic with respect to the major axes of the structure) occur in the delamination mode.

Delamination fracture is an important failure mode in laminated composite materials. It is associated with failure of the weakest components of the composite material—the matrix and the fiber-matrix interface. Delamination is often the limiting factor in the use of composites for structural applications, particularly for long-term use, where thermal cycling, mechanical fatigue, and environmental effects such as moisture are present. Delamination can occur at stress concentrations such as holes, material discontinuities such as ply drop-offs, or joints. The study of delamination fracture, therefore, has drawn increasing attention.[6]

Most of the current effort has been in measurement of the delamination fracture toughness under a variety of loading and environmental conditions, as well as in the analysis of delamination using continuum laminated plate theory. Little work has been done thus far in developing design guidelines for using delamination fracture toughness in the design of composite structures, particularly for spacecraft structures.

Delamination Fracture Mechanics

In the study of delamination fracture toughness of composite laminates, linear elastic fracture mechanics (LEFM) is used. This approach adequately describes delamination fracture propagation since it fulfills the two major requirements for the applicability of LEFM to composites: The crack must lie in a plane of material symmetry and propagate in a collinear or self-similar fashion.[5] The delamination fracture toughness can be readily measured for a variety of spacecraft structural composite materials. Details of the technique, including the specimen design and the testing method, can be found in Ref. 7.

For mode I crack propagation, LEFM gives the crack energy release rate[8]

$$G_{Ic} = (P_c^2/2w)\,(\mathrm{d}C/\mathrm{d}a) \tag{1}$$

For the double cantilever beam (DCB) specimen, the compliance can be estimated using a strength of materials approach. Approximating the DCB specimen as two cantilevered beams, and assuming that the contribution to the compliance from the crack tip is independent of the crack length,

$$C = \delta/P = 2a^3/3EI \tag{2}$$

Substituting in Eq. (1)

$$G_{Ic} = P_c^2 a^2/EIw \tag{3}$$

In the foregoing, implicit assumptions are made that shear deformations are small, and that viscoelastic effects are restricted to a small zone at the crack tip. Whitney et al. have derived an expression for the DCB specimen for the crack energy release rate in which shear deformation is included[9]; however, the shear deformation effect can be neglected for crack lengths that are large relative to the thickness of the specimen, which is usually the case.

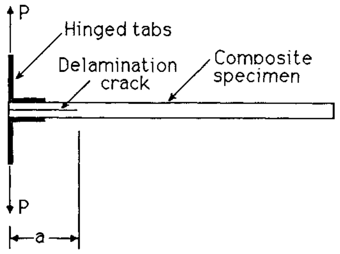

Fig. 1 Delamination fracture specimen.

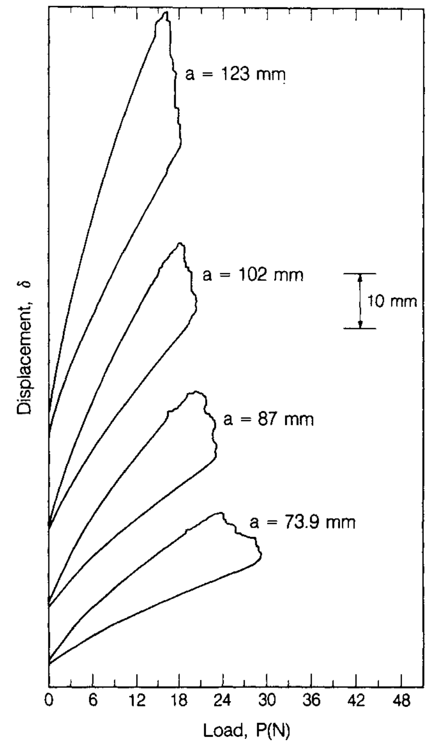

Fig. 2 Crack opening displacement vs load for incremental crack propagation during mode I delamination.

Viscoelastic effects in delamination have been studied by Devitt et al.,[10] for mode I delamination fracture and were found to be negligible if viscoelasticity were to be restricted to a small region near the crack tip. In the present investigation, the errors involved in neglecting shear deformation and viscoelastic effects are small within the context of the experimental techniques used, and can be neglected.

For the purpose of establishing delamination fracture toughness values for typical composite materials used in spacecraft structures, the delamination fracture toughnesses of these materials were measured.

Delamination Fracture Toughness Measurement

Specimen Design and Materials

The double cantilever beam specimen is used to characterize mode I behavior. The mode I specimen shown in Fig. 1 is a DCB specimen with bonded hinged metal tabs attached at one end for the application of a load perpendicular to the interlaminar layer. Initiation of delamination along the interlaminar layer is accomplished by the insertion of a piece of polytetrafluoroethylene (PTFE) tape at the desired interface at the time of fabrication.

For the present investigation, laminates were constructed from woven and unidirectional Amoco T-300/Fiberite 934 graphite pre-preg and from woven DuPont Kevlar/Fiberite

934 aramid pre-preg. All of these materials pass the outgassing test for spacecraft materials (volatile condensible materials <0.1% and total mass loss <1.0%), and are approved for spacecraft use.

The unidirectional material was layed up with all plies at 0 deg, and the woven material was layed up in the 0/90 direction. In addition, laminates were prepared with two epoxy films at the mid-plane, with the PTFE tape between the films. This was done to measure the in situ delamination fracture toughness of the matrix. All of the laminates were cured in an autoclave at 170°C for 4 h at 0.7 MPa pressure. The resulting fiber volume was found to be 65 ± 1%.

Test Method

The specimens were loaded at constant crosshead velocity in an Instron machine. Initially, before crack length measurements were made, each specimen was loaded to grow the imbedded crack (initiated from the PTFE tape) to an initial crack approximately 30 mm long. The specimen was loaded at constant crosshead velocity until the crack just began to propagate. The specimen was then unloaded and the process repeated until the crack grew to about 75% of the specimen length. The crack length was measured during the reloading cycle just before crack propagation began.

Figure 2 shows typical crack opening displacement vs the applied load responses. In order to evaluate the effects of loading rate on delamination fracture toughness, two crosshead speeds were used: 0.083 and 0.83 mm/s. Scanning electron microscopy was used to observe the fracture surfaces and identify the failure modes.

Results and Discussion

Table 1 shows crack energy release rates for the materials tested. Also shown are data for unidirectional glass-reinforced epoxy reported by Devitt et al.,[10] and for unidirectional graphite-reinforced epoxy (T-300/Fiberite 934) reported by Garg and Ishai.[11]

It is evident that increasing the loading rate by a factor of 10 results in an increase in the crack energy release rate by about 15% in the three types of laminates tested and the epoxy layer. Thus, there is a rate-dependent effect in delamination fracture.

There is also an increase in the crack energy release rate for a woven composite compared to a unidirectionally reinforced composite for graphite composites (the only material tested in both woven and unwoven configurations). This can be explained by observations of the fracture surface. Scanning electron microscopy of the graphite-epoxy delaminated fracture surface shows little fiber participation in the fracture process. The delamination crack propagates in the matrix between the fibers and along the fiber-matrix interface. The fracture surface for the T-300/934 composites is characterized by clean fiber surfaces exposed by the crack, indicating low interfacial energies.

Similar fracture morphology was reported by Richards-Frandsen and Naerheim for delamination of graphite-epoxy composites.[12] The fracture surface showed the characteristic exposed clean fiber surface with traces of removed fibers separated by the fractured epoxy matrix surface, which was characterized by hackles.

In mode I crack propagation in unidirectionally reinforced composites, locally, at the crack tip, the crack has to work around the imbedded fibers as it propagates. Locally, therefore, the fracture process involves both mode I and mode III components for unidirectional laminates in which the "macroscopic" opening mode crack front propagates parallel to the fiber direction.

For woven laminates, because of the waviness of the weave, the crack front is more convoluted than in the unidirectionally reinforced material, and, locally, all three fracture modes are present. This observation, and the fact that the fracture surface is larger on a local (microscopic) level than the projected area that is used to calculate the fracture toughness, could account for the increase in the fracture toughness of woven composites over that of unidirectional composites. For graphite-epoxy composites, the woven geometry increases the crack energy release rate by a factor of about 2.5 over the unidirectional material.

The woven Kevlar/934 laminate showed a higher fracture toughness than the woven graphite laminates. Once again, scanning electron microscopy of the fracture surface shows that, like the graphite composites, there is very little evidence of matrix adhesion to the fiber surfaces exposed by the delamination crack. However, one observes the presence of several broken fibrils. The well-known tendency of aramid fibers to fibrillate, i.e., split longitudinally into fibrils, may account for the higher delamination fracture energy observed in aramid composites. A possible mechanism is that, as the opening mode delamination crack propagates, some fibrils are separated and bridge the crack. As the crack propagates, these fibrils have to fracture, thus introducing a contribution to the delamination fracture energy by the fibers in addition to the normal matrix and interfacial failures.

Finally, the crack energy release rates for the epoxy alone, measured in situ by using molded-in thin resin films, are useful since the results can be used to estimate the relative contributions to the overall delamination fracture toughness from the fiber-matrix interface and the matrix alone. Bulk resin fracture toughness data are not particularly useful, since crack propagation in the bulk occurs without the constraints provided by the fibers in a composite. Thus, the fracture toughness in the bulk resin can be much larger than the toughness of the matrix within the composite.

Application to Spacecraft Structures

Composite spacecraft structures are generally designed for high stiffness and dimensional stability. To meet these requirements, such structures require a high degree of anistotropy. This is because, in order to obtain an overall coefficient of thermal expansion (CTE) close to zero, there must be a large percentage of 0 deg high modulus graphite-epoxy plies. Typi-

Table 1 Mode I crack energy release rates (J/m²)

	Loading rate, mm/s	
	0.083	0.83
Unidirectional Thornel 300/934	90	—
graphite-epoxy	103 (Ref. 6)[a]	—
Woven Thornel 300/934 graphite-epoxy	232	266
Woven Kevlar/934	330	382
In situ epoxy layer	164	191
Unidirectional glass-epoxy	525–1020 (Ref. 5)	—

[a] Reference data shown for comparison.

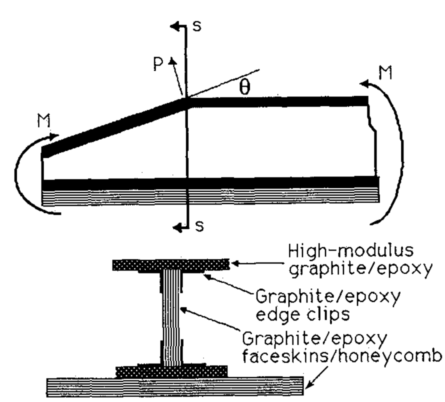

Fig. 3 Section of a typical stiffened composite spacecraft structure showing the small out-of-plane load P produced by the external moments M.

cally, there may be 75–85% of 0 deg plies consisting of high modulus fibers such as Celanese Corporation's GY-80 or Union Carbide's P-75. Such construction results in laminates with low transverse strength that are susceptible to delamination following microcracking during thermal cycling. Even curing stresses can be large enough to cause delamination.

For parabolic antennas, especially the relatively flat platelike geometry of offset C-band reflectors, stiffeners are necessary in order to meet the launch vibration environment. For the INTELSAT V and INTELSAT VI C-band reflectors, for example, I-beam stiffeners are employed to stiffen a thin composite sandwich shell.[13] A typical geometry of the stiffener is shown in Fig. 3, in which the reflector shell is of sandwich construction with composite faceskins. The I-beam stiffeners consist of flanges containing predominantly high-modulus unidirectional graphite fiber bonded to a shear web of sandwich construction.

Such construction is very efficient for producing high stiffness with minimum weight. By matching the longitudinal laminate CTE of the flanges of the stiffener with that of the shell, it is possible to produce a structure that is, theoretically, dimensionally stable. Such structures are usually well analyzed for the in-plane loads (arising from the launch vibration environment) that they are designed to accommodate.

Small out-of-plane loads, such as the bending moment shown in Fig. 3, can, however, cause delamination of such a construction. Such loads can arise from the vibration environment, from handling, or from supporting minor attachment brackets for thermal blankets, for example. Incorporating delamination fracture criteria in defining the minimum margins of safety of the entire structure will ensure that such unanticipated failures do not occur.

One can analyze this structure by assuming that a delamination flaw exists within the plies constituting the high-modulus stiffener (Fig. 3). The application of a bending moment results in an out-of-plane load P that will tend to propagate the flaw. When this happens, the contribution of the stiffener to the structure is lost and a large change in the fundamental frequency of the structure can result. It is important to note that this sort of failure is initiated by a fracture mode in which fibers do not fail. Thus, stress analysis that takes into account in-plane loading only (in which fiber-direction strains are calculated and compared with allowables) cannot predict such a failure. A fracture-control approach that screens a structure for just this type of crack propagation, however, will detect such a problem.

In analyzing this geometry using linear elastic fracture mechanics, one obtains a relationship relating the critical bending moment M_c that will result in catastrophic propagation of a critical flaw of length $2a_c$ (the complete expression is derived in Ref. 14):

$$M_c = a_c^{-1} f\,(G_{Ic}, \text{beam geometry}, \theta, E_1, E_2, \nu_{12}, G_{12}) \qquad (4)$$

Figure 4 is a plot of the curve relating the critical bending moment necessary for catastrophic delamination crack propagation as a function of the critical delamination half-crack size, a_c, for two matrices of differing delamination fracture toughnesses. This curve was calculated for a specific geometry and, for $\theta = 15$ deg. Derivation of the equation used to generate this curve is cumbersome and is published elsewhere.[14] Knowing the critical crack size (as defined by the specific inspection technique used), the critical bending moment necessary for the catastrophic propagation of this flaw can be determined using Eq. (4), or a curve such as shown in Fig. 4. Since this is also a function of the fracture toughness G_{Ic} and the beam angle θ, such an analysis can define the degree of improvement required, if necessary, in the fracture toughness. Alternatively, one can modify the beam geometry for a given detectable flaw size and fracture toughness.

For completeness, one needs also to conduct a crack propagation analysis to determine the maximum tolerable initial flaw size if catastrophic propagation is not to occur for a given life (in load cycles). This feature can be incorporated in a fracture-control plan that includes delamination in composite materials by using an equation of the type[8]

$$\mathrm{d}a/\mathrm{d}N = C\,(\Delta G_I)^n \qquad (5)$$

where C and n are empirical constants in an experimentally determined curve relating the crack growth per load cycle $\mathrm{d}a/\mathrm{d}N$ to the applied crack energy release rate ΔG_I.

Crack propagation is a mandatory component of the current fracture-control plan for metals and should be included in a proposed plan for composite structures. It is not discussed in detail in the present paper because of space limitations. For delamination in composite laminates, however, the exponent n is typically high (in the range 3–4.5), while for metals it lies in the range 1.5–2.[15,16] This means that, for composites, the slope of the log $\mathrm{d}a/\mathrm{d}N$ vs log ΔG_I curve is steep, i.e., the difference between a "threshold" fracture toughness value and the catastrophic value G_{Ic}, is small. Thus, there is a small (though nonconservative) error in using G_{Ic} and ignoring crack propagation considerations.

A complete fracture-control plan should also include minimum strain criteria, nondestructive techniques to be used, and a flowchart for defining the acceptability of a structure based on inspection and analytical criteria. These aspects are described in detail elsewhere.[14]

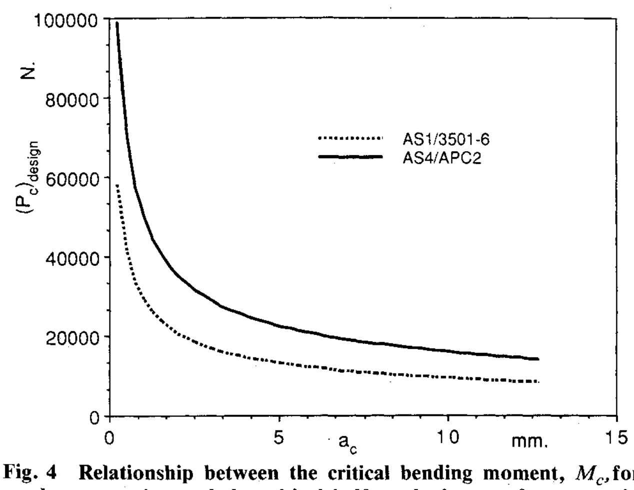

Fig. 4 Relationship between the critical bending moment, M_c, for crack propagation and the critical half-crack size, a_c, for two resin matrices.

Conclusions

The elements of an approach toward a fracture-control plan for composite spacecraft structures is presented. The plan is based on delamination, which is an important failure mode in engineering composite laminates. Thus, the plan envisages the use of delamination fracture toughness as a criterion for catastrophic crack propagation.

The opening mode (mode I) delamination fracture toughness was measured for unidirectional and woven graphite-epoxy laminates, and for aramid-epoxy laminates. In addition, the in situ mode I crack energy release rate for the epoxy matrix was also measured by delaminating thin epoxy films. The results show that the mode I delamination fracture toughness of graphite composites is small relative to glass composites. Measured values of 90 J/m^2 were obtained for unidirectional T-300/934 compared with 525-1020 J/m^2 for glass-epoxy.

Woven graphite composites exhibited a delamination fracture toughness of about 2.5 times that of the unidirectionally reinforced material. This effect has been attributed to the waviness of the weave and the associated greater fracture surface area, and to the presence, locally, of all three crack propagation modes.

The fracture surface of the graphite-epoxy laminates was characterized by clean exposed fiber surfaces indicating poor interfacial fracture toughness. The aramid composites also showed relatively clean fiber surfaces; however, some fibrillation was observed. The fibrillation could account for the higher fracture toughness of the aramid composite over the graphite-epoxy material because it introduces some fiber participation in the delamination process.

The effect of loading rate was to increase the crack energy release rate by about 15% for a tenfold increase in the loading rate. This relative increase was obtained for all of the materials tested.

Analysis of a typical high-stiffness construction using high-modulus graphite fiber/epoxy shows that the incorporation of delamination fracture mechanics analysis in the design of composite structures will allow for the assessment of pre-existing flaws. Such an approach is inherently fail-safe since the presence of interlaminar flaws can be detected by current nondestructive evaluation techniques.

The incorporation of delamination fracture toughness considerations in the design of composite spacecraft structures is strongly recommended as a means toward improving the reliability of such highly anisotropic structures.

Acknowledgment

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