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DESIGN CRITERIA FOR X-CRV HONEYCOMB PANELS - A PRELIMINARY STUDY

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ABSTRACT

The objective of this project is to perform the first step in developing structural design criteria for composite sandwich panels that are to be used in the aeroshell of the crew return vehicle (X-CRV). The preliminary concept includes a simplified method for assessing the allowable strength in the laminate material. Ultimately, it is intended that the design criteria be extended to address the global response of the vehicle. This task will require execution of a test program as outlined in the recommendation section of this report.

The aeroshell of the X-CRV is comprised of composite sandwich panels consisting of fiberite face sheets and a phenolic honeycomb core. The function of the crew return vehicle is to enable the safe return of injured or ill crewpersons from space station, the evacuation of crew in case of emergency or the return of crew if an orbiter is not available. A significant objective of the X-CRV project is to demonstrate that this vehicle can be designed, built and operated at lower cost and at a significantly faster development time. Development time can be reduced by driving out issues in both structural design and manufacturing concurrently. This means that structural design and analysis progresses in conjunction with manufacturing and testing.

Preliminary tests results on laminate coupons are presented in the report. Based on these results a method for detection material failure in the material is presented. In the long term, extrapolation of coupon data to large scale structures may be inadequate. Test coupons used to develop failure criteria at the material scale are typically small when compared to the overall structure. Their inherent small size indicates that the material failure criteria can be used to predict localized failure of the structure, however, it can not be used to predict failure for all failure modes. Some failure modes occur only when the structure or one of its sub-components are studied as a whole. Conversely, localized failure may not indicate failure of the structure as a whole and the amount of reserve capacity, if any, should be assessed.

To develop a complete design criteria experimental studies of the sandwich panel are needed. Only then can a conservative and accurate design criteria be developed. This criteria should include effects of flaws and defects, and environmental factors such as temperature and moisture. Preliminary results presented in this report suggest that a simplified analysis can be used to predict the strength of a laminate. Testing for environmental effects have yet to be included in this work. The so called "rogue flaw test" appears to be a promising method for assessing the effect of a defect in a laminate. This method fits in quite well with the philosophy of achieving a damage tolerant design.

1.0 INTRODUCTION

The aeroshell of the crew return vehicle (X-CRV) vehicle is comprised of composite sandwich panels consisting of fiberite face sheets and a phenolic honeycomb core. A first step in developing structural design criteria for these sandwich panels is the primary focus of this report. It includes a simplified method for assessing the allowable strength in the laminate material. Ultimately it is intended that the design criteria be extended to address the global response of the vehicle. This task will require execution of a test program as outlined in the recommendation section of this report.

The function of the crew return vehicle is to enable the safe return of injured or ill crewpersons from space station, the evacuation of crew in case of emergency or the return of crew if an orbiter is not available. The shape of the X-CRV is based upon the USAF/Martin X-24A lifting body vehicle. A significant objective of the X-CRV project is to demonstrate that this vehicle can be designed, built and operated at lower cost and at a significantly faster development time.

Substantial cost and weight savings can be achieved in aerospace vehicles through the use of composite materials. For example, the NASA/DoD advanced composites technologies program has a projected goal of 30%-50% weight reduction and a 20%-25% cost savings when compared to a baseline aluminum wing and fuselage structure of a commercial vehicle in production (Smith et. al, 1995). The predicted savings are to be achieved by a synergism of innovative design and manufacturing techniques. Similar efficiencies are also anticipated for an aerospace vehicle such as the X-CRV. To this end, the experimental version of the crew return vehicle, the X-CRV, uses a composite sandwich as the structural element for the outer mold line (OML). One of the primary advantages in construction of this type over aluminum structures is that the part size can be relatively large. The cost savings accrue due to a reduction of labor required to fabricate and assemble these parts. Development time can be reduced by driving out issues in both structural design and manufacturing concurrently. Furthermore, structural design and analysis progresses in conjunction with manufacturing and testing.

1.1 Development of Design Criteria

The focus of this report is to summarize a preliminary assessment of a simple and operational criteria that can be used in the structural design of the X-CRV composite outer aeroshell. A structural design criteria can first be developed based upon classical mechanics failure criteria. These criteria are used to predict failure of the material locally. Preventing material failure is a first step in assuring the structural integrity.

Test coupons used to develop failure criteria at the material scale are typically small when compared to the overall structure. This is especially true in a vehicle such as the X-CRV. When these test coupons are fabricated they are effectively cut away from a larger sample of the base material. Their inherent small size indicates that the material failure criteria can be used to predict localized failure of the structure, however, it can not be used to predict failure for all failure modes. Some failure modes occur only when the structure or one of its subcomponents are studied as a whole. These failure modes and are not predicted by coupons tests. Conversely, localized failure may not indicate failure of the structure as a whole and the amount of reserve capacity, if any, should be assessed.

In the long term, extrapolation of coupon data to large scale structures may be inadequate. Experimental verification of the failure criteria using components representative of the global structural response is imperative. Testing of two-way panels is recommended for use in the verification process. The validity and conservativeness of the design criteria can then be assessed with confidence and results of panel tests can be correlated to finite element models of the structure.

In the end, the design criteria employed should be one that is reasonably accurate and operationally simple. In choosing this criteria a realistic look at the design process is required. Ideally, a more complicated engineering solution is typically thought of as being more accurate. In an ideal situation the more complicated solution may be the engineers choice because intuition tells us that it will yield a more optimum design at the expense of some design time. Realistically, however, designs are subject to change caused by updates to the desired service, cost concerns, and to correct for errors. Feeding this information back into a more complicated analytical model may not be desirable. It will increase the likelihood of errors and delay the design. Therefore, the choice of a reasonably accurate, conservative, simpler criteria is better, especially for preliminary design.

1.2 Failure Criteria

The rationale behind mechanics failure criteria is that data from relatively simple tests can be used to predict the failure of a material as it is subjected to more complex states of stress and strain. Failure criteria are based largely upon observation of test data and can be thought of as operational rather than mechanistic. Failure criteria are empirical expressions that are related to the material failure process.

In polymer composites, better correlation between predictions and experiments have been observed when failure criteria is applied to the laminate rather than on a ply-by-ply basis (Norr et. al , 1983). In other words, a more accurate evaluation of the structural response will be determined from tests and analysis performed at the macroscopic scale. The ultimate strength of a laminate can be predicted using the relevant strength data for a single ply. This, however, is a relatively complex problem. It is more appropriate to rely upon test data for laminate strength values. The basic material tests for the laminate consists of standard coupon tests that are typically used to quantify material properties. These test are done on a routine basis and data from these tests are input into the failure model. The same holds true for evaluation of a composite sandwich and it will be more accurate to evaluate this sandwich material at a scale large enough to represent the behavior of the structural component.

There are numerous material failure criteria that have been developed in the past. Rowlands (1985) gives a comprehensive summary of a multitude of criteria. Two of the criteria used frequently in the analysis of composite materials, the maximum strain criteria (St. Venant) and the Tsai-Wu criteria, will be emphasized in this report.

For the most part, failure criteria all agree in failure prediction when the material is subjected to simple states of stress such as uniaxial tension, uniaxial compression or pure shear. They differ when the effects of combined loading are to be predicted. Furthermore, failure criteria are to be used with caution because they will not predict all failure modes, especially those that occur at the sub-component or component scale.

1.2.1 Tsai-Wu failure criteria - The Tsai-Wu failure criteria for orthortopic lamina can be written as follows:

$$F_1\sigma_x + F_2\sigma_y + F_{11}\sigma_x^2 + 2F_{12}\sigma_x\sigma_y + F_{22}\sigma_y^2 + F_{66}\sigma_s^2 = FI$$

where

FI= the failure index

$$F_1 = \frac{1}{X_t} - \frac{1}{X_c}, \qquad F_2 = \frac{1}{Y_t} - \frac{1}{Y_c}, \qquad F_{11} = \frac{1}{X_t X_c}$$

$$F_{22} = \frac{1}{Y_t Y_c}, \qquad F_{66} = \frac{1}{S^2}, \quad F_{12} = -\frac{1}{2}\sqrt{F_{11}F_{22}}$$

$X_t =$ tensile strength in X - material direction

$Y_t =$ tensile strength in Y - material direction

$X_c =$ compressive strength in X - material direction

$X_t =$ compressive strength in Y - material direction

$S =$ shear strength referenced to the X - Y plane

This criteria uses linear and quadratic stress terms. The non-linearity of this criteria indicates that the failure index will not be directly proportionally to the applied loads.

<u>1.2.2 Maximum Strain Failure Criteria</u>- Using this theory failure is predicted when the applied strain in a principal material direction exceeds the maximum allowable principal strain in that direction. Mathematically speaking, failure occurs when any of the following inequalities are violated.

$$\varepsilon_x^+ < \varepsilon u_{xt}\ ,\ \ \varepsilon_x^- > -\varepsilon u_{xc}\ ,\ \ \varepsilon_y^+ < \varepsilon u_{yt}\ ,\ \ \varepsilon_y^- > -\varepsilon u_{yc}\ ,\ \ |\gamma_{xy}| < \gamma u_{xy}$$

where

$\varepsilon u_{xt}, \varepsilon u_{xc}, \varepsilon u_{yt}, \varepsilon u_{yc}, \gamma u_{xy}$ are the ultimate strains,

t indicates tension and c compression

$\varepsilon_x^+, \varepsilon_x^-, \varepsilon_y^+, \varepsilon_y^-$ are the applied strains

In this case the resulting strains are directly proportional to the applied loads. A margin of safety can be determined by direct scaling.

<u>1.2.3 Ply-by-Ply Analysis</u>- In design of composites strain compatibility is assumed between adjacent elements of the composites. In other words, the strain between two adjacent infinitesimal coupons of materials is the same. A composite in bending will have a linear strain distribution through the thickness. However, the state of stress between two adjacent elements is not the same when the two elements have different compliance's. The stress state will be discontinuous through the thickness. This makes a ply-by-ply stress analysis of a composite laminate operationally tedious.

1.3 Damage Tolerant Criteria

In developing values for allowable material behavior a damage tolerant design is desired where a component will maintain its structural integrity while a given defect is present. Damage tolerant design relies on accepting that damage will occur, on implementing a system to detect damage, and on a design where adequate strength is maintained in the damaged structure. Damage tolerance in composites is complex primarily due to the non-homogeneous nature of the material. Behavior of most carbon fiber reinforced composites is nearly linear up to failure and the failures are sudden. In manufacturing of composites a larger number of defects may exist when compared to metallic structures.

Influence of an allowable defect can be incorporated into the design process. Defects in composites can occur due to manufacturing preparation and production, machining, processing and of the assembly of the component. Some of these defects include voids, delaminations, disbonds, foreign object inclusions, resin starved or rich areas, incomplete resin cure, misaligned fiber orientation, fiber gaps, wrinkled layers, and poor surface conditions. In sandwich panels additional defects may include poor core splice, disbonds from facesheets, crushed core and core gaps. Assembly defects may result from scratches gouges, incorrect drilling of holes and tool impact damage. The decision to repair or reject the component will be based upon the size and nature of the defect and its influence on the structural performance. Accurate methods of evaluating the effect of the defect on the structure and detection of the defect in the first place are required.

Undesirable structural response due to accumulation of damage under hysteresis loading, impact and fatigue should be mitigated. Damage due to impact depends upon the energy level of the impact. High energy impacts will typically produce visible damage and procedures can be set in action to mitigate the effects of this type of damage on the structure. Low energy impact damage can be detrimental to the long-term life of the structure and this damage is often difficult to detect. In composites damage is expected to occur at quite low impact energy levels. This is especially true for sandwich panels. Thickening of the facesheets will help to lessen damage. Impact from an object such as a toolbox dropped from 1 foot may produce significant damage in the composite structure.

Tension fatigue is not normally a problem in composites. Tests have shown that for specimens with holes tension fatigue life is not a major concern. Specimens loaded to 90% of their ultimate tensile strength have achieved fatigue life of over 1 million cycles (Hoskin and Baker, 1986). On the other hand, compression specimens have shown a reduced fatigue life similar to that of metals. One million cycles was not achieved unless the compressive stress was limited to less than 40 percent of the ultimate strength.

Damage tolerant design takes into account the type and size of defect that may be present in the structure. Once the size and nature of the allowable defect is established, its effect on the structure can be accounted for in the design process. Figure 1 shows the effect of such a hole on the failure strength. In this case a 0.25 inch hole will reduce the failure strength to 60% of its unflawed value. A crack of a 0.25 inch length will have a similar effect on the tensile strength of the material.

One method for reducing the design allowables in composites design is the rogue flaw test. In this test a 1/4 inch diameter hole (flaw) is incorporated into a 1 inch wide tensile specimen. In doing so, stress concentrations will occur around the flaw and the ultimate strength and strain will be reduced by a substantial amount. This results in allowables that can be used at the material scale. Other damage tolerant criteria can be developed following a similar philosophy.

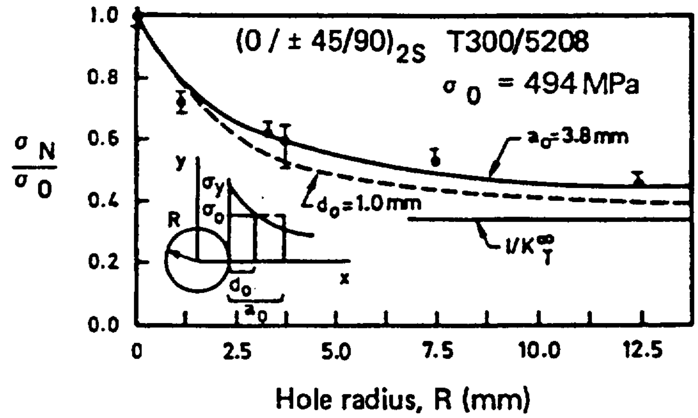

Figure 1 - Effect of Hole on Tensile Strength of a graphite/epoxy Laminate, predicted and experimental (from Hoskin and Baker, 1986).

1.4 Influence of Environmental Conditions

Temperature extremes and moisture, especially in combination can significantly influence the structural properties of a composite. These effects must be considered in design. Tests should be conducted on representative samples to assess the environmental influence. Moisture is absorbed and desorbed in composites by a diffusion process. This process is relatively slow and is highly temperature dependent. Moisture can reduce the glass transition temperature and in combination with high temperature it can reduce strength. The cyanate resin, for example, undergoes a modulus decrease of 23% and a flexural strength decrease of 30% at 325° F/wet conditions when compared to dry room temperature conditions. Compressive properties of fiber dominated laminates are typically more affected by these conditions since a modulus reduction of the resin increases the likelihood of fiber buckling at a lower stress.

2. APPROXIMATE DEFINITION OF ALLOWABLE STRENGTH

The purpose of this section is to describe the theoretical considerations in development of allowable strength values for the Fiberite/Phenolic Honeycomb sandwich panels. The sandwich panels are made from two primary constituents, namely, the face sheets made of Fiberite 3454-2AJ tape and the honeycomb core made of Hexel Flexcore HRP/F35-4.5 phenolic core. For purposes of this report an example configuration consisting of 1.5 inch thick Flexcore and face sheets with 14 layers [45,-45,90,0,90,0,90]s will be discussed in regard to defining the allowable strength.

2.1 Constituent Materials

The fiberite ply consists of a fiber that is graphite based and a Cyanate Resin 954-2A. For analysis purposes these constituent materials were assumed to be isotropic and their baseline properties are given in Table 1.

A theoretical ply material was constructed using the graphite fiber and cyanate resin properties listed above. The ply calculation facility of the IDEAS program was used for this purpose. In these computations the parameters were chosen as summarized in Table 2.

TABLE 1A- PROPERTIES OF FIBER AND RESIN MATERIALS

	Graphite Fiber	Cyanate Resin
Longitudinal Modulus, Ex, (psi)	78 x 10^6	44 x 10^6
Poisson's Ratio, ν_{xy}	0.25	0.38
Shear Modulus, Gxy, (psi)	31.2 x 10^6	15.9 x 10^6
Ultimate Tensile Strength,(psi)	583 x 10^3	10.0 x 10^3
Ultimate Tensile Strain, μstrain	7455*	25,900
Specific Gravity	1.91	1.24

* Approximate based on ideally brittle behavior.

TABLE 1B- PROPERTIES OF CORE MATERIAL

	Flexcore
Compressive Modulus, (psi)	49 x 10^3
Minimum Stabilized Compressive Strength (psi)	470
Shear Modulus, X direction (psi)	22 x 10^3
Shear Modulus, Y direction (psi)	12 x 10^3
Minimum Shear Strength , X direction (psi)	220
Minimum Shear Strength , Y direction (psi)	110
Specific Gravity	0.0721

TABLE 2 - INPUT FOR IDEAS PLY PROPERTIES CALCULATION

Matrix	Cyanate
Fiber	Graphite
matrix vol. fraction, Vm	0.44
fiber vol. fraction, Vf	0.56
reinforcing factors	no
theory for elastic constants	Mechanics of Materials
theory for thermal properties	Levin

2.2 Fiberite Ply Properties

Calculated ply properties were determined using the ply definition module of the IDEAS computer code. These properties can be compared to material data given by Fiberite. The ply properties for a unidirectional fiberite sheet are summarized in Table 3. The last column of this table gives nominal values that will be used in a demonstration model presented in Section 2.3. The nominal strain values assume elastic-brittle behavior and are computed by taking the nominal measured failure strength divided by the elastic modulus.

TABLE 3 - SUMMARY OF FIBERITE PLY PROPERTIES

	Calculated IDEAS	Fiberite data sheet	Nominal
Longitudinal Modulus, Ex (psi)	43.9×10^6	45.5×10^6 (T) 42.7×10^6 (C)	*44.1×10^6
Transverse Modulus, Ey (psi)	1.12×10^6		1.12×10^6
Poisson's Ratio, Vxy	0.307		0.307
Shear Modulus, Gxy (psi)	0.36×10^6		0.36×10^6
Long. Tensile Strength, Xt (psi)		310.8×10^3	310.8×10^3
Long. Comp. Strength, Xc (psi)		130.4×10^3	130.4×10^3
Trans. Tensile Strength, Yt (psi)		3.4×10^3	3.4×10^3
Trans. Comp. Strength, Yc (psi)		1.5×10^3	1.5×10^3
Shear Strength, S (psi)		10.5×10^3	10.5×10^3
Long. ult. tensile strain, εux_t ,ustrain			7048
Long. ult. comp. strain, εux_c ,ustrain			2957
Trans. ult. tensile strain, εuy_t ,ustrain			3036
Trans. ult. comp. strain. εuy_c ,ustrain			1339
Ultimate shear strain, γu ,ustrain			29,176

* *Tensile and compressive properties were averaged*

2.3 Fiberite Laminate Face Sheet

An example face sheet can be constructed using the ply material summarized in Table 3 for demonstrating a simplified analysis procedure. This example is of a 14 ply laminate with a layer thickness of 0.0048 inches. The lay-up for this laminate is [45,-45,90,0,90,0,90]s. In this development the primary material axes of the laminate are designated by x and y subscripts whereas direction of applied stress and strain are designated by a 1 or 2 subscript.

Stiffness properties were calculated using the laminate definition module of the IDEAS program. Strength estimations can be made for this laminate based on the assumption that when the laminate is subjected to a unidirectional state of stress along a primary material axis, only the fibers oriented along the direction of the stress will contribute to the ultimate strength. This gives what is called the Simplified Analysis Limit. A classic failure criteria such as Tsai-Wu or Maximum Strain can be used to predict strengths under combined states of stress assuming that the laminate acts as an ideally orthotropic material.

2.3.1 Parameters for TSAI-WU Strength analysis.

In a TSAI-WU analysis five laminate strength parameters are required, namely, Xt, Yt, Xc, Yc, and S. These parameters can be determined using five separate tests, two tension tests, two compression tests and a shear test. Without test data they can be approximated by making the assumption stated above as the Simplified Analysis Limit.

Tensile Strength Parameters - This test can be thought of as a uniaxial tension test with the material oriented in the zero or ninety degree direction. In the longitudinal, 0, direction there are a total of 4 plies and there are a total of 6 plies at 90. The stress at failure is calculated as:

$$Xt = (4/14)*310.8 \times 10^3 = 88.8 \times 10^3 \text{ psi}$$

$$Yt = (6/14)*310.8 \times 10^3 = 133.2 \times 10^3 \text{ psi}$$

Compressive strength - This test can be thought of as a compression test, global buckling prevented, with the material oriented in the zero or ninety degree direction. The stress at failure is calculated as:

$$Xc = (4/14)*130.4 \times 10^3 = 37.2 \times 10^3 \text{ psi}$$

$$Yc = (6/14)*130.4 \times 10^3 = 55.8 \times 10^3 \text{ psi}$$

Shear strength - The shear strength can be thought of as applying a state of pure shear, $\tau_{xy} = \sigma$, on an element that does not buckle. In pure shear a rotation of 45 degrees will yield a state of stress that can be defined as follows:

$$\sigma_1 = \sigma \qquad \sigma_2 = -\sigma \qquad \tau_{12} = 0$$

For this laminate σ_1 will be oriented along the +45 degree axis of the laminate and will be in tension. The stress σ_2 will be oriented along the -45 degree axis of the laminate and will be in compression. The controlling factor will be the compressive strength of the -45 degree plies. There are two plies to resist this stress. Accordingly, the shear strength can be approximated as:

$$S = (2/14)^* \ 130.4 \times 10^3 = 18.6 \times 10^3 \text{ psi}$$

The tensile , compressive and shear strengths listed above can be used in the Tsai-Wu expression across the laminate. When this is done the coefficients in the Tsai-Wu expression become.

$$F_1 = -1.56 \times 10^{-5}, \ F_2 = -1.04 \times 10^{-5}, \ F_{11} = 3.03 \times 10^{-10},$$
$$F_{22} = 1.35 \times 10^{-10}, \ F_{12} = -1.0 \times 10^{-10}, \ F_{66} = 2.88 \times 10^{-9}$$

Tensile and Compressive Tests at 45 degrees

Once these parameters are chosen other states of stress can be analyzed to asses the validity of this hypothesis. For starters, a theoretical uniaxial tensile test at an axis of +45 degrees can be applied to the laminate. This results in a state of stress along the primary material axes as follows:

$$\sigma_x = 0.5^*\sigma \qquad \sigma_y = 0.5^*\sigma \qquad \tau_{xy} = -0.5^*\sigma$$

The stress, σ, that brings the Failure Index to 1.0 will be σ=45.2 x 10^3 psi

Failure can also be predicted using the same criteria as done in the calculation of the tension strength parameter. Failure occurs when the plies along the tension direction fail. Since there are two +45 degree plies the tensile strength is predicted as:

$$S_{45t} = (2/14)^* \ 310.8 \times 10^3 = 44.4 \times 10^3 \text{ psi}$$

For practical purposes these two values are nearly the same.

Unfortunately, the same does not hold true for a compression test at 45 degrees. In this case the compressive strength is predicted as:

$$S_{45c} = (2/14)^* \ 130.4 \times 10^3 = -18.6 \times 10^3 \text{ psi}$$

The compressive stress, σ, that brings the Failure Index to 1.0 is σ=-28.5 x 10^3 psi

2.3.2 Parameters for Maximum Strain Analysis

Parameters for use in the maximum strain analysis will be developed based upon properties of the laminate. The effective modulus of the laminate calculated based upon classical lamination theory are as follows:

$$E_x = 15.8 \times 10^6$$
$$E_y = 21.7 \times 10^6$$
$$G_{xy} = 3.42 \times 10^6$$

The maximum strain can be computed based upon the stress at failure determined by the Simplified Limit Analysis. This analysis will achieve the same stress level at failure during simple tension, compression and shear loading. For the demonstration laminate the strain parameters are as follows:

$$\varepsilon u_{xt} = \frac{88.8 \times 10^3}{15.8 \times 10^6} = 5620\mu\varepsilon, \quad \varepsilon u_{yt} = \frac{133.2 \times 10^3}{21.7 \times 10^6} = 6138\mu\varepsilon, \quad \varepsilon u_{xc} = \frac{37.2 \times 10^3}{15.8 \times 10^6} = 2354\mu\varepsilon,$$

$$\varepsilon u_{yc} = \frac{55.8 \times 10^3}{21.7 \times 10^6} = 2571\mu\varepsilon, \quad \gamma u_{xy} = \frac{18.6 \times 10^3}{3.42 \times 10^6} = 5444\mu\varepsilon$$

Table 4 provides a summary of properties for the [45,-45,90,0,90,0,90]s laminate.

TABLE 4 - SUMMARY OF THE LAMINATE PROPERTIES

Laminate	[45,-45,90,0,90,0,90]s	Long. Tensile Strength, Xt (psi)	44.4×10^3
Long. Modulus, Ex (psi)	15.8×10^6	Long. Comp. Strength, Xc (psi)	18.6×10^3
Trans. Modulus, Ey (psi)	21.7×10^6	Trans. Tensile Strength, Yt (psi)	66.6×10^3
Poisson's Ratio, Vxy	0.149	Trans. Comp. Strength, Yc (psi)	27.9×10^3
Poisson's Ratio, Vyz	0.149	Shear Strength, S (psi)	18.6×10^3
Poisson's Ratio, Vzx	0.149	Long. ult. tensile strain, eu_{xt} ,ustrain	5620
Shear Modulus, Gxy (psi)	3.42×10^6	Long. ult. comp. strain, eu_{xc} ,ustrain	2354
Shear Modulus, Gyz (psi)	0.342×10^6	Trans. ult. tensile strain, eu_{yt} ,ustrain	6138
Shear Modulus, Gzx (psi)	0.347×10^6	Trans. ult. comp. strain. eu_{yc} ,ustrain	2571
Laminate Thickness, in.	0.0672	Ultimate shear strain, γu ,ustrain	5444

3. TENSION TESTS OF FIBERITE LAMINATES.

The validity of the simplified approach needs to be verified by testing. This section presents a preliminary testing program which can be thought of as a first step in the validation process. More tests are required. In this phase tests were conducted on two different lay-ups of laminates similar to what would be a facesheet in a honeycomb panel. These facesheets were made of the Fiberite 3454-2AJ tape and the lay-ups were as follows:

8 Ply - [0,90,45,-45]s and 16 Ply - [90,0,0,90,0,60,-60,0]s

Stiffness properties of these laminates can be computed theoretically as given in Table 5.

TABLE 5 - MODULUS AND POISSON'S RATIO, 8-PLY AND 16-PLY LAMINATES.

	8 - Ply	16 - Ply
Longitudinal Modulus, Ex (psi)	15.2×10^6	22.8×10^6
Transverse Modulus, Ey (psi)	15.2×10^6	17.6×10^6
Effective Modulus at 45 Degrees, (psi)	15.2×10^6	7.9×10^6
Poisons Ratio, Vxy	0.33	0.13
Shear Modulus, Gxy (psi)	5.72×10^6	2.37×10^6

Strength and strain parameters can be calculated using lamination theory and the simplified method presented in Section 2. These values are presented in Table 6.

TABLE 6 - STRENGTH AND MAXIMUM STRAIN PARAMETERS.

	8 Ply Simplified | Theory	16 Ply Simplified | Theory
Long. Tensile Strength, Xt (psi)	77.7×10^3 | 107.3×10^3	155.4×10^3 | 160.9×10^3
Long. Comp. Strength, Xc (psi)	32.6×10^3	65.2×10^3
Trans. Tensile Strength, Yt (psi)	77.7×10^3 | 107.3×10^3	77.7×10^3 | 122.7×10^3
Trans. Comp. Strength, Yc (psi)	32.6×10^3	35.6×10^3
Shear Strength, S (psi)	32.6×10^3	15.2×10^3
Long. ult. tensile strain, εux_t ,ustrain	5112	6815
Long. ult. comp. strain, εux_c ,ustrain	2145	2857
Trans. ult. tensile strain, εuy_t ,ustrain	5112	4415
Trans. ult. comp. strain. εuy_c ,ustrain	2145	2023
Ultimate shear strain, γu ,ustrain	5699	6414

Tensile test specimens were cut from a 12" by 12" sheet of the laminate material. The specimens were 8" in length and 0.5" in width. Three sets of five were cut from each sheet for a total of thirty samples. The first set was cut in the direction of the plies in the outer layer. The second and third sets were cut at 90 degrees and 45 degrees to this direction . Specimens were designated as follows:

*P-@@-0#

where * represents the number of plies (8 or 16), @ represents the angle of cut relative to the direction of the top ply layer, and # represents the specimen number in the set. For example, specimen 16P-90-03 is a 16 ply specimen cut at 90 degrees to the top ply (this is the laminate's strong or X-direction) and is the third specimen in the series. Holes of 1/8" diameter were drilled in the center of the fourth and fifth specimen of each set.

One of the objectives of this preliminary study was to develop some experience in testing these materials. Obviously, due to the small sample size this series of tests does not represent a statistically significant sample and the reader is cautioned to not rely too heavily on these results. The intention was to gain some initial insight into the behavior of these laminates and to develop a starting point for further study.

It was found that testing of straight specimens is more of an art than a science. The desired failure in a specimen is one where it fails in the gage area. Several gripping techniques were attempted. ASTM D3039 (1995) recommends use of an emery cloth between the grip and the specimen for straight laminates and this method was selected for these tests. The response of the specimen is sensitive to grip method grip pressure and alignment of the specimen in the machine. Use of a fixed grip tends to increase the problem of alignment, therefore, ball and socket grip connections are recommended at top and bottom.

Even with all of these precautions taken a majority of the specimens failed near to the grip area. Intuition tells us that this will most likely be the case. Failure in a specimen will initiate at a flaw or due to a stress concentration in the specimen. The likelihood is small that a flaw will be more severe than the stress concentration near the grips.

Other recommended methods for gripping are use of tabs or fabrication of tapered specimens. The tabs will help mitigate damage to the outer layers of the laminate and may help to reduce the concentration of stress at the grips. The effectiveness of the tabs can only be assessed by testing. Fabrication of tapered specimens is possible. Care should be taken to make the taper gentle enough so that stress concentrations are not excessive at the taper. This would only move the problem from one region to another. If a comprehensive laminate testing program is to be undertaken an attempt at developing a tapered specimen should be made. If this is not successful or practical it should be followed by studying the feasibility of using tabs.

In testing of simulated flawed specimens with holes the gripping problems do not exist. These test results are expected to be more consistent, and they were. Failure will virtually always occur through the flawed region.

3.1 Test Results

Results of the Fiberite laminate tests are summarized in Table 7, including the results of both the straight specimens and the specimen with holes. The computations of modulus of elasticity, E, and ultimate stress were based upon the nominal thickness. The last column of this table provides a ratio of the strength of the laminate to the strength determined by the simplified method. For the off axis specimens cut at 45 degrees (8P-45 and 16P-45) the ratio was based upon the limiting value of stress determined by the maximum strain criteria using the parameters presented in Table 6.

Referring to the predicted modulus of elasticity presented in Table 5 and the experimentally determined modulus given in Table 7 it can be seen that this parameter can be predicted with good accuracy using classic lamination theory. The average percent difference in this prediction is -0.3% for the 8-Ply laminate, -3% for 16P-0, 6% for 16P-90 and -0.2% for 16P-45.

In the 16-ply laminate tested on its strong axis (16P-90, Figure 3) one-half of the plies in this case are oriented in the strong direction. The cross plies being at 60 degrees indicates that a relatively small amount of load sharing will be contributed by these plies. Also, this causes the simplified analysis limit to be relatively close to the lamination theory limit. Difficulties were observed in achieving the strength indicated by the simplified analysis limit as none of the three tests achieved this limit. In the other direction (16P-0, Figure 4) the simplified analysis limit was achieved in two of the three tests and the lower strength in the third tests was due to problems at the grips. In this case substantial load sharing occurs by the cross plies which are located at an angle of thirty degrees. The 8-ply material (Figures 5-6) is a quasi-isotropic material and one would expect the same behavior in either direction. In this lay-up significant load sharing exists due to the 45 degree plies. Four of the six tests achieved the simplified analysis limit.

In the off-axis tests at 45 degrees (8P-45 and 16P-45) the strength can be predicted using the parameters presented in Table 6. Table 8 gives a summary of these predicted strengths using both the maximum strain and the Tsai-Wu criteria. The maximum strain criteria gives a more conservative prediction of the strength in this orientation. In the tests, the strength of 16-ply material exceeded these predictions as indicated by a ratio greater than 1 in Table 7. For the 8-ply material it may appear, at first glance, that both the maximum strain criteria and the Tsai-Wu criteria are overly conservative and that the strength should reach the same value as in the X and Y material direction (77.7 ksi) since this material is quasi-isotropic. However, at 45 degrees this material makeup looks like a [-45,45,0,90]s laminate and Crossman (1983) has shown that this stacking sequence is prone to delamination and generally will exhibit a substantially lower ultimate strength.

TABLE 7 - RESULTS OF THE LAMINATE TESTS.

Specimen	Nom. t	W	t	Peak Load	Ult. Stress	Ult. Strain	E	Ratio
	in.	in.	in.	lbs.	psi	ustrain	Msi	
16P-0-01	0.0768	0.5102	0.0815	3675	93,796	5500	16.7	1.21
16P-0-02	0.0768	0.5020	0.0810	3748	97,215	5500	17.3	1.25
16P-0-03	0.0768	0.5023	0.0802	2850	73,874	4600	17.0	0.95
16P-0-04	0.0768	0.5043	0.0818	2263	58,426	3600	16.1	0.75
16P-0-05	0.0768	0.5045	0.0818	2255	58,200	3700	15.5	0.75
16P-90-01	0.0768	0.5013	0.0797	4803	124,745	4800	26.0	0.80
16P-90-02	0.0768	0.5025	0.0800	5267	136,479	6000	23.3	0.88
16P-90-03	0.0768	0.5042	0.0797	5745	148,373	6400	23.2	0.95
16P-90-04	0.0768	0.5007	0.0803	3160	82,182	4200	19.8	0.53
16P-90-05	0.0768	0.5043	0.0802	3231	83,418	4100	20.4	0.54
16P-45-01	0.0768	0.5058	0.0802	1554	40,002	5000	8.2	1.31
16P-45-02	0.0768	0.5040	0.0807	1371	35,420	4600	7.8	1.16
16P-45-03	0.0768	0.5033	0.0803	1447	37,433	4900	7.8	1.22
16P-45-04	0.0768	0.5030	0.0805	1166	30,183	3700	8.3	0.99
16P-45-05	0.0768	0.5018	0.0808	1222	31,707	4400	7.6	1.04
8P-0-01	0.0384	0.5027	0.0402	1595	82,632	5500	15.1	1.06
8P-0-02	0.0384	0.5022	0.0403	1391	72,135	4600	15.6	0.93
8P-0-03	0.0384	0.5033	0.0402	1143	59,137	4100	15.1	0.76
8P-0-04	0.0384	0.5023	0.0403	1124	58,270	4200	13.9	0.75
8P-0-05	0.0384	0.5035	0.0405	1147	59,324	4400	13.7	0.76
8P-90-01	0.0384	0.5015	0.0407	1610	83,603	5500	15.4	1.08
8P-90-02	0.0384	0.5027	0.0407	1596	82,684	5500	15.0	1.06
8P-90-03	0.0384	0.5022	0.0403	1711	88,730	5600	15.9	1.14
8P-90-04	0.0384	0.5023	0.0403	954	49,457	3600	11.8	0.64
8P-90-05	0.0384	0.5033	0.0403	992	51,325	3500	14.8	0.66
8P-45-01	0.0384	0.5037	0.0402	1191	61,580	4500	14.5	0.95
8P-45-02	0.0384	0.5027	0.0402	1198	62,065	4300	14.0	0.95
8P-45-03	0.0384	0.5027	0.0403	1201	62,220	4200	15.8	0.96
8P-45-04	0.0384	0.5022	0.0400	928	48,125	3700	14.2	0.74
8P-45-05	0.0384	0.5023	0.0400	840	43,547	3600	12.4	0.67

TABLE 8 - STRENGTH PREDICTION AT 45 DEGREES.

Designation	Maximum Strain Criteria	Tsai-Wu Criteria
8P-45 (psi)	65.1×10^3	72.9×10^3
16P-45 (psi)	30.6×10^3	34.4×10^3

The rogue flaw tests of specimens with holes generally achieved 53%-70% (see Table 7) of the value predicted by simplified analysis. In the tests of 16P-45 this percentage was greater primarily due to an underprediction of the strength in this direction by the maximum strain criteria. Theoretically, the stress concentration factor for this hole is grater than 3. The test results indicate a strength reduction factor less than 2 thereby indicating a significant redistrubution of stress around the hole prior to failure.

4. RECOMMENDATIONS

The principal outcome of this preliminary study is to recommend an extensive testing program for the XCRV composite sandwich panels. These recommendations are summarized as follows:

1. Perform an exhaustive search of the literature for experimental results which may pertain to this material.

2. Perform a comprehensive test program to evaluate the structural response of the X-CRV sandwich panels at the sub-component and component scale. Correlate these test results to finite element analysis. Include environmental effects, defects and impact.

3. Supplement the laminate coupon tests with additional tests of the laminate and sandwich core system. These tests will help to interpret the results of 2) above. They should be made of constituents used in the above study.

4. Develop a comprehensive design criteria based upon the results of the above studies.

5. CONCLUSIONS

A comprehensive design criteria can not be developed from material coupon tests alone. Structural response from tests at the sub-component and component level are required. This is especially true in composites and composite sandwich panels where theoretical strength analysis is complex. There are a multiplicity of failure modes that depend upon structural geometry, load path and practical issues such as manufacturing techniques.

Experimental study of the sandwich panel as a whole is needed so that the global structural response be understood. Only then can a conservative and accurate design criteria be developed. The design procedure should be operationally simple and related to the methods of the analytical tool that will be used in design, typically finite element analysis. This criteria should include provisions for detecting any and all possible failure modes, both local and global. Conversely, localized material failure may not indicate failure of the structure or the component as a whole. Component testing will result in an assessment of reserve capacity.

A rational approach is required for preliminary design which is often performed in the absence of test results. Preliminary results presented in this report suggest that a simplified analysis can be used to predict the strength of a laminate coupon. Care should be exercised in choosing strength and strain parameters especially when there is not a significant number of plies in the laminate at an angle of 45 degrees or less. Environmental effects have yet to be included in this work. The so called "rogue flaw test" appears to be a promising method for assessing the effect of a defect in a laminate. This method fits in quite well with the philosophy of achieving a damage tolerant design.

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