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A Study of Flexible Composites for Expandable Space Structures

Stephen J. Scotti
Langley Research Center, Hampton, Virginia

March 2016

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Stephen J. Scotti
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National Aeronautics and
Space Administration

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Abstract:

Payload volume for launch vehicles is a critical constraint that impacts spacecraft design. Deployment mechanisms, such as those used for solar arrays and antennas, are approaches that have successfully accommodated this constraint, however, providing pressurized volumes that can be packaged compactly at launch and expanded in space is still a challenge. One approach that has been under development for many years is to utilize softgoods – woven fabric for straps, cloth, and with appropriate coatings, bladders – to provide this expandable pressure vessel capability. The mechanics of woven structure is complicated by a response that is nonlinear and often nonrepeatable due to the discrete nature of the woven fiber architecture. This complexity reduces engineering confidence to reliably design and certify these structures, which increases costs due to increased requirements for system testing. The present study explores flexible composite materials systems as an alternative to the heritage softgoods approach. Materials were obtained from vendors who utilize flexible composites for non-aerospace products to determine some initial physical and mechanical properties of the materials. Uniaxial mechanical testing was performed to obtain the stress-strain response of the flexible composites and the failure behavior. A failure criterion was developed from the data, and a space habitat application was used to provide an estimate of the relative performance of flexible composites compared to the heritage softgoods approach. Initial results are promising with a 25% mass savings estimated for the flexible composite solution.

Introduction:

Inflatable space structures are an important technology for human exploration in space. The salient feature that makes them attractive is their ability to be stowed into a small payload volume at launch, and then be inflated into a larger structure in space or at a planetary destination. Two uses of inflatable structures are habitats for crews in space and on planetary surfaces, and high-drag decelerators to facilitate the entry of a payload into a planetary atmosphere. The heritage development path for structural softgoods in NASA habitation applications is based on the Mars Transhab effort of the 1990s (ref. 1). This technology has been adopted and developed by Bigalow Aerospace (ref. 2). A small, Bigalow-developed, inflatable module will be attached to the International Space Station (ISS) Node 3 Aft Port in 2016. The use of inflatable structures for entry system decelerators was first studied in the 1960s and a recent embodiment is the Inflatable Reentry Vehicle Experiment (IRVE), a stacked-toroid blunted cone Hypersonic Inflatable Atmospheric Decelerator (HIAD) geometry (ref. 3). The structural capability in both these inflatable concepts is derived from pressurized membranes woven from high-strength fibers such as Kevlar ™ or Vectran ™. These flexible membranes react the internal pressure loads, and either applied coatings or non-structural bladders prevent leakage of the pressurizing gas. (ref. 4)

Though the concept of an inflatable structure is simple, the present implementation using woven structural membranes produces multiscale material effects that impact the mechanical behavior of the structure as illustrated in Figure 1.

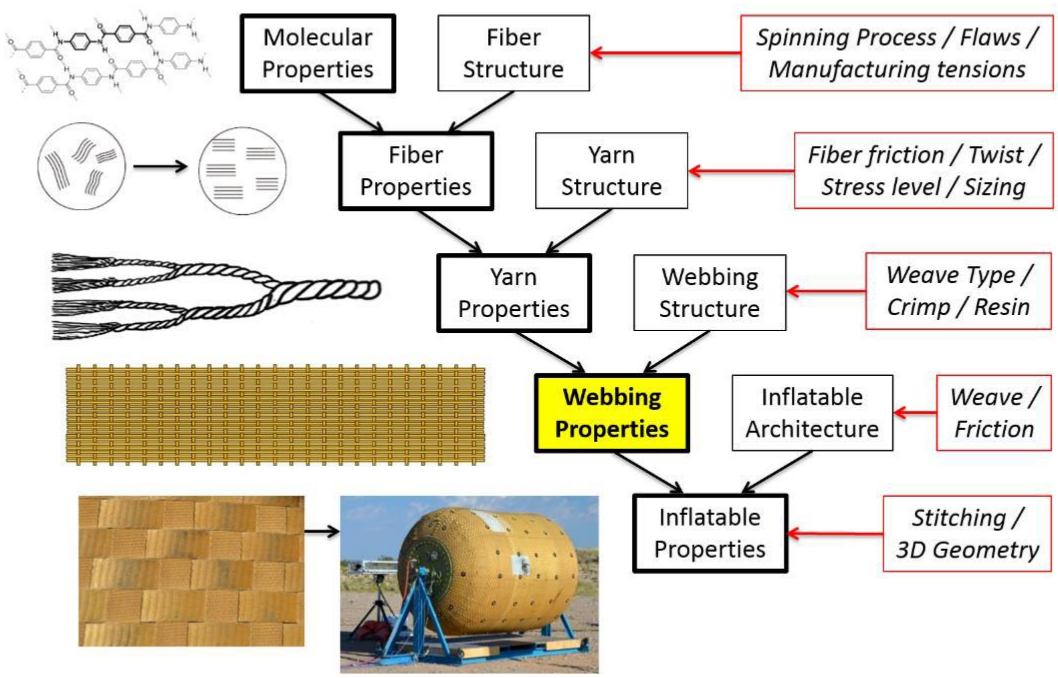

Figure 1 (taken from ref. 4) - Multiscale material effects of heritage woven flexible structure.

This multiscale response complicates the understanding of the response and failure of the woven structure. For example, the load (stress)-strain characteristics of the woven materials are nonlinear as illustrated in Figure 2. In addition to the nonlinearity, they exhibit a large scatter in the response and failure loads between samples as well as load history dependence when a single sample is cycled. Because of these complications, NASA has required a factor of safety for woven structures that is double that normally used for metallic or composite materials (Note: ref 5. indicates that "due to the larger manufacturing tolerances, load sharing, and loading uncertainties involved with fabrics, the Federal Aviation Administration (Airship Design Criteria, FAA-P-81 00-2) requires that all inflatable airships (blimps) be designed to FOS=4.0 for ultimate. Therefore, the TransHab team imposed this same requirement on the restraint layer design.”). This requirement is codified in NASA STD-5001B (ref. 6).

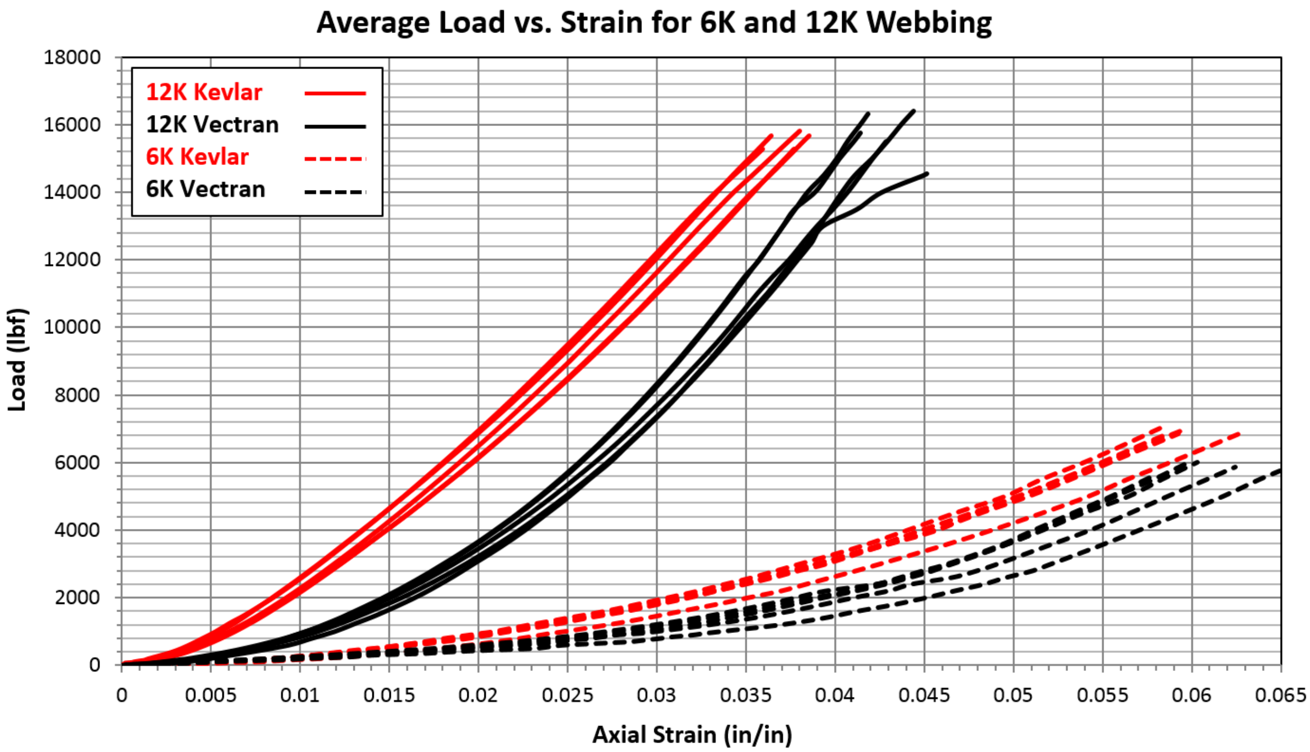

Figure 2 (taken from ref. 4) - Average Load vs. Strain curves for woven Vectran and Kevlar webbings.

In addition to (and partially as a result of) the larger factor of safety required for a woven, flexible structure, it will be stressed in a very nonlinear portion of its material stress-strain response – for example, 12K Vectran in figure 2 will operate at about 0.011 strain level and the response has high curvature at this condition – that complicates the analysis and verification of the load transfer within complex flexible structures. Thus, it will be necessary to perform significantly more testing in the development and verification of a flexible, woven structure because its analysis will be less reliable due to both the nonlinearity and the scatter in properties.

A new class of materials that could decrease the engineering development time and cost, and also enable mass reductions from the heritage approach of woven, flexible structures is explored in the present paper. Flexible composites, made from thin plies of high-strength fibers and flexible matrix resins, have been developed for sails of racing sailboats (see ref. 7) and they are envisioned to have significant advantages in their application to flexible space structures. Flexible composite space structures would be designed and fabricated identically to the approach used for state-of-the-art rigid structural composites – 1) a design that optimizes the composite laminate layups that is tailored to the loads in the deployed configuration would be developed using standard composite design approaches, 2) a mold or tool of the deployed configuration would be fabricated and the thin-ply, flexible prepreg would be placed on the tool according to the design, 3) vacuum bagging would be applied and the composite thermally cured, 4) the bagging and tool would then be removed and the flexible composite structure is available for integration into the rest of the system. There would be variations of this approach depending on the method of prepreg placement (e.g., hand or automated tape placement) and whether an autoclave is used during the cure stage. The advantages a composite approach have over the woven softgoods approach are that composite design, analysis and fabrication are relatively mature, so the engineering and manufacturing processes are well in hand. Because the material response of a composite is linear and reliably repeatable, the composite flexible structure design can utilize a lower factor of safety than woven softgoods (which should reduce mass though the addition of a resin matrix would temper this reduction in mass), and there is a much improved ability to model a composite structure compared to woven softgoods (which should reduce development and certification testing requirements and costs). As the flexible composite technology matures, there may be additional mass benefits obtained from applying composite bonding technology to the integration of flexible composite components within a space system. It may even be possible to fabricate rigid composite components integrated with flexible composites using the same prepreg placement and cure process (e.g., create rigid elements in the structure with a simple change of the type of prepreg tape being used in the layup process). Such an approach would eliminate the heavy and cumbersome mechanical attachment interfaces to rigid elements in the heritage softgoods design (e.g., ref. 8), simplify the assembly of the full structural system, and ensure a hermetic seal of the interface between the rigid and flexible layers of the structural wall. However, exploration of this integrated concept is left for future research efforts.

The present paper explores the capabilities of flexible composites obtained from two sources, and their applicability to a NASA mission. The use of the heritage woven softgoods approach is discussed for a NASA habitat application to provide a set of properties to compare with the flexible composite approach. The flexible composite materials under consideration are described, and the architectures of the laminates tested are presented. The test procedure for the static testing of the samples is discussed, and the test results presented. The paper concludes with a discussion of the applicability of the flexible composite technology to NASA missions, and the suggested next steps in their development.

Material Systems Studied:

Two sources for flexible composite laminates were evaluated in the present study. Samples prefixed with “A” were prepared from materials provided by North Sails company, and samples prefixed with “B” were prepared from material provided by the Army Research Lab – Aberdeen.

Material A – North Sails Nevada, the provider of material A laminates, have several proprietary prepregs that are produced internally, and used to fabricate sails for racing sailboats (ref. 7). They can produce custom prepregs and fabricate composite laminates using various blends of carbon, Aramid, and Ultra-high-molecular-weight polyethylene (UHMWPE) fibers in a polyester resin matrix. An “off-the-shelf” material having a 60% fiber mass density and consisting of Aramid and UHMWPE fibers in a ratio of 2-to-1 (by mass) was used for all the material A laminates. Five custom laminates shown in table 1 were provided from which specimens were extracted for physical and mechanical property measurements. (For future reference, the ply thickness and density obtained by averaging over laminates A-001 through A-004 are 0.0012 in. and 0.0309 lb_m/in^3, respectively.) The first 4 laminates were selected to be similar to those used for rigid composites – quasi-isotropic for A-001 and A-002, and a 2-to-1 ratio of 0-degree to 90-degree plies for laminates A-003 and A-004. One can see that the 0-degree plies are single ply blocks for A-001 and A-002, two ply blocks for A-004, and three ply blocks for A-003. The last laminate layup has 6 plies in each 0-degree block, and was selected to evaluate whether a highly tailored laminate could replace a braided cable. Photos of some of the 3-inch square samples (for A-001 through A-004) used to obtain the average thickness and density in table 1 are shown in figure 3. Fiber orientation for the outer ply is vertical and a criss-cross pattern of white threads is also visible. These polyester threads are a non-structural “carrier scrim” that improves handling of the ultra-thin prepreg during fabrication. The laminates are flexible, however, they are somewhat “boardy”, and figure 4 shows the result when an A-003 laminate is folded to a 90 degree angle across the 0 direction and over ~1/4 in. radius. The laminate recovers to have a small permanent set, and several permanent surface “creases” were formed in the radius as seen in figure 4. Several of the samples tested to obtain mechanical properties included similar surface wrinkles, however, no significant reduction in strength due to the creases was observed.

Table 1 – Material A composite laminates.

Laminate Designation	A-001	A-002	A-003	A-004	A-005
Ply Count	32	16	35	18	30
Ply orientations	$[0/-45/90/45]_{4S}$	$[0/-45/90/45]_{2S}$	$[(0)_3/45/-45/(0)_3/(90)_3/45/-45/(0)_3/90/0.5*90]_S$	$[(0)_2/45/-45/(90)_2/(0)_2/45]_S$	$[(0)_6/45/-45/(0)_6/45/-45/(0)_6/-45/45/(0)_6]$
Laminate thickness (in.)	0.0407	0.0196	0.0397	0.0209	0.0327
Density (lb_m/in^3)	0.0282	0.0307	0.0329	0.0317	0.0356

Material B – The Army Research Lab (ARL) develops lightweight personal protection equipment (helmets, body armor, etc.) and has significant experience in fabrication and testing of Ultra-high molecular weight polyethylene (UHMWPE) composite materials for this purpose. To investigate whether the material used for the Army applications would be useful for flexible composite needs of NASA as well, ARL supplied material B laminates for this study. Material B laminates were fabricated by ARL from the Dyneema (DSM Dyneema LLC) HB80 cross-plied "prepreg" using a lamination process they had previously developed. The HB80 material uses Dyneema SK76 fibers in a polyurethane matrix. The nominal fiber volume is 82% and the HB80 material laminated by ARL is crossplied with a [0/90/0/90] layup. Four layers of the HB80 prepreg were used to produce two 24-inch-square quasi-isotropic laminates with layup $[0/90/0/90/45/-45/45/-45]_s$. The measured average thickness of these laminates was 0.0265 in., and the density was 0.030 lb_m/in^3. Note that layups for laminates A-001 and A-002 are the ones most similar to the material B laminate. However, even though the material B laminate is much thinner than A-001 (0.0265 in. vs 0.0407 in.), it was anticipated to be stronger due to the higher fiber volume fraction as well as being 100% UHMWPE fiber which is stronger than the aramid fiber that is the preponderate reinforcement in material A. Material B laminates were also "boardy" and much more so than material A laminates. Figure 4 shows the permanent deformation in a strip of material B laminate after bending 90 degrees around a ~1/4 in. radius. There was little recovery from the 90 degree bending, and white striations were formed along the crease. These striations are believed to be matrix damage due to matrix cracking and delamination.

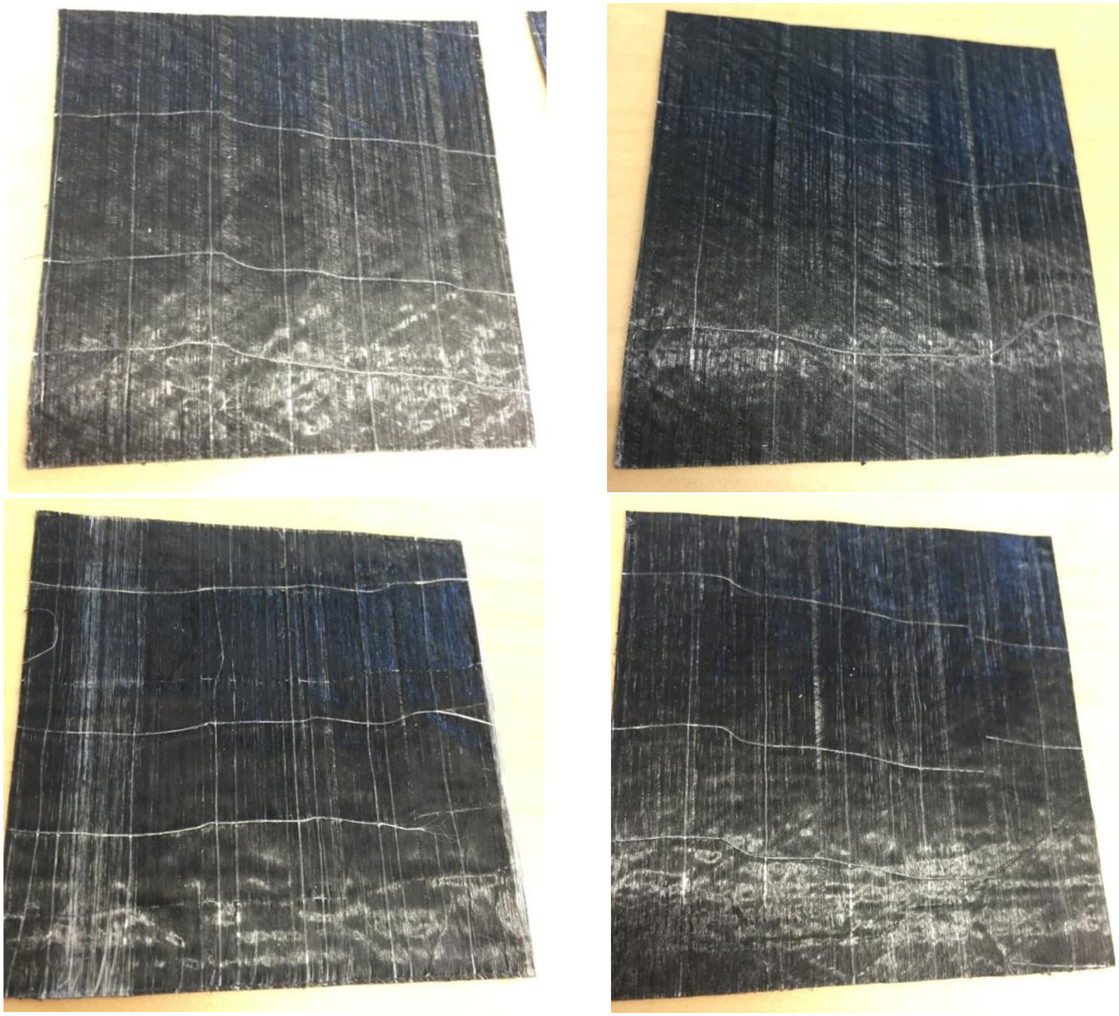

Figure 3 – Samples of material A [North Sails] laminates A-001 (upper left), A-002 (upper right), A-003 (lower left), and A-004 (lower right).

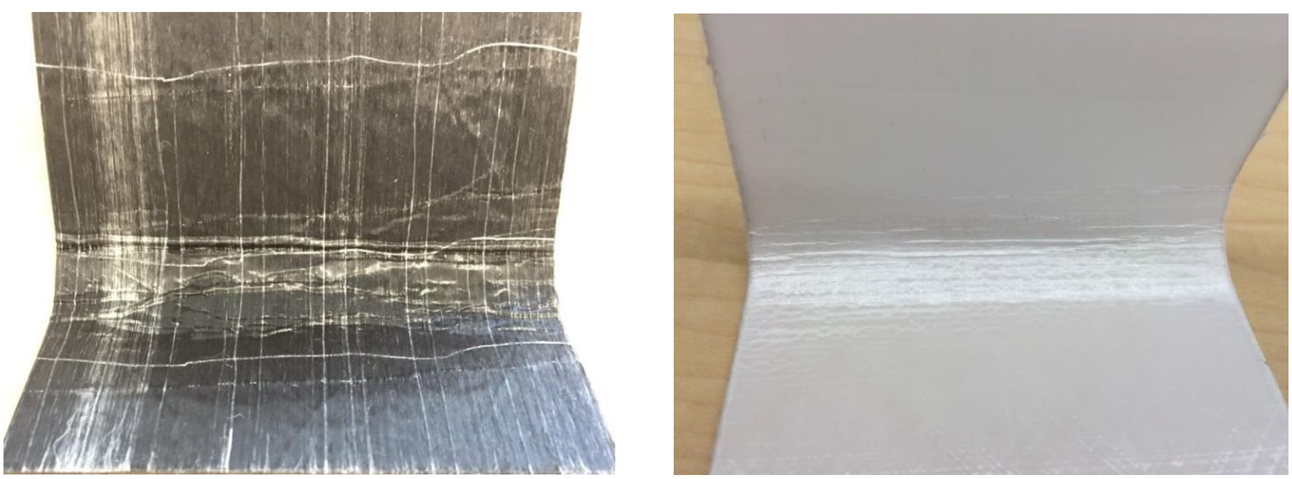

Figure 4 – Samples of material A [North Sails] laminate A-003 (left) and the material B [ARL] laminate (right) after bending to ~1/4 in. radius.

Test Procedure:

Uniaxial mechanical tests were performed to measure the strength of the flexible composite materials A and B. For specific tests, full field strain measurements were obtained using digital image correlation techniques (VIC-3D, ref. 9) in order to obtain stiffness data as well as to determine if there are localized nonuniformities in the material response. Two specimen configurations were investigated: straps were cut from the cured panels having either constant width or a dogbone-shape. Most of the tests were done in the 0-degree laminate direction, however, for samples of laminates A-003 and A-004, tests were also performed in the 90-degree laminate direction. Figure 5 shows the two sample configurations mounted in a universal testing machine. The strap test samples (figure 5 left) must be fairly long because they are threaded in a serpentine manner in the double-pin fixture that uses friction to grip the specimen. The dogbone samples utilizing hydraulic grips (figure 5 right) can be much shorter, and the hydraulic grip pressure used was approximately 700 psi. In this figure, the samples are shown with a speckle pattern that is used in the digital image correlation measurements. The specimen width in the gauge section was 1 inch for both configurations. Midway through the strap test series using double-pin grips, the specimen configuration was modified to a dogbone-in-the-strap configuration to eliminate failure at the grips. The strap testing using double-pin grips was replaced with a dogbone specimen configuration using hydraulic grips for the 90-degree tests for laminates A-003 and A-004, and for the material B samples because this approach was more efficient in material usage, and it was the only testing configuration possible for the shorter material B samples.

Most of the tests were performed measuring only load and machine displacement, however, for each laminate type there was at least 1 test performed using digital image correlation so that stiffness parameters and global material response could be obtained. However, there were no digital image correlation measurements used for the 90-degree laminate direction tests. All the tests were performed in displacement control with a fairly slow loading rate, 0.1 inch/minute.

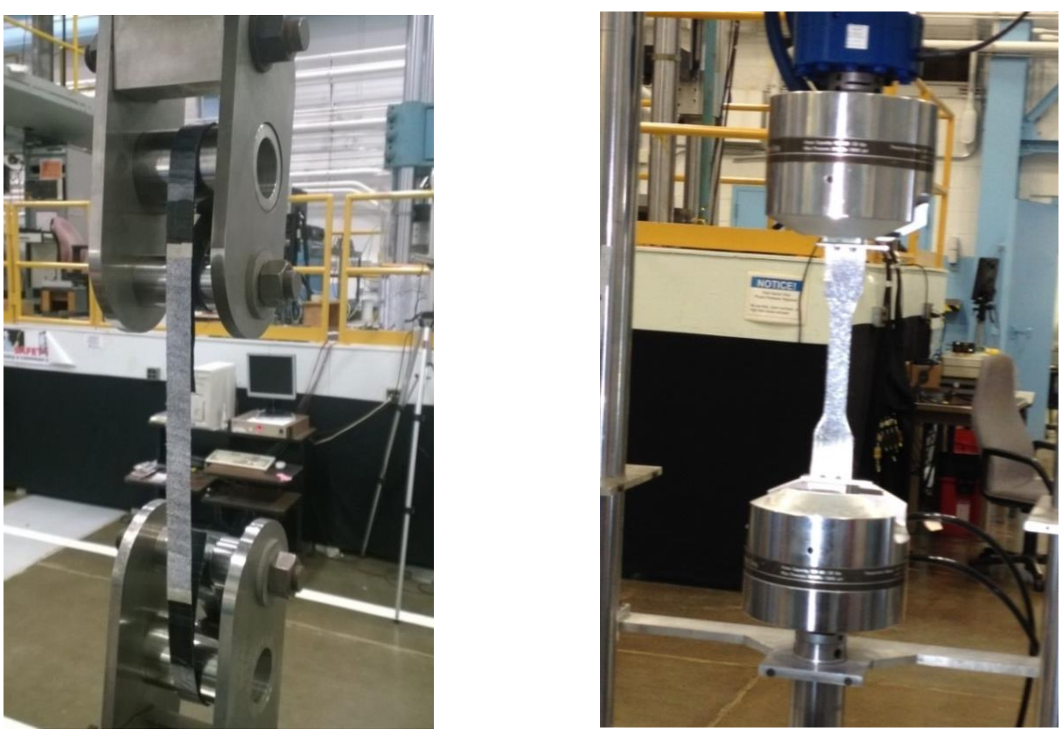

Figure 5. Strap (constant width) specimen in Mil-T-87130 double-pin grip fixture (left) and dogbone specimen in hydraulic grip fixture (right) installed in universal testing machine for tension tests.

Results:

Test Failure Summary - Table 2 summarizes the tests performed showing the specimen type, the failure load, and the strain from the digital image correlation at failure load (for the material A samples only).

Table 2. Test Result Summary Table.

Sample Number	Laminate	Specimen/Fixture type	VIC-3D	Failure load (lb)	Failure Strain
A-001-28	A-001	Dogbone/Double-pin	Yes	949.6	0.0192
A-001-1	A-001	Strap/Double-pin	No	856.1	-
A-001-2	A-001	Strap/Double-pin	No	909.6	-
A-001-3	A-001	Strap/Double-pin	No	840.1	-
A-001-4	A-001	Strap/Double-pin	No	895.9	-
A-001-6	A-001	Dogbone/Double-pin	No	950.8	-
A-002-18	A-002	Dogbone/Double-pin	Yes	545.2	0.0207
A-002-19	A-002	Dogbone/Double-pin	Yes	485.7	0.0184
A-002-20	A-002	Dogbone/Double-pin	Yes	493.5	0.0210
A-002-1	A-002	Strap/Double-pin	No	466.5	-
A-002-2	A-002	Strap/Double-pin	No	438.8	-
A-002-3	A-002	Strap/Double-pin	No	493.7	-
A-002-4	A-002	Strap/Double-pin	No	506.8	-
A-002-5	A-002	Dogbone/Double-pin	No	546.4	-
A-002-6	A-002	Dogbone/Double-pin	No	488.5	-
A-002-7	A-002	Dogbone/Double-pin	No	496.7	-
A-003-29	A-003	Dogbone/Double-pin	Yes	1727.8	0.0170
A-003-1	A-003	Strap/Double-pin	No	1926.5	-
A-003-2	A-003	Strap/Double-pin	No	1855.2	-
A-003-3	A-003	Strap/Double-pin	No	1745.4	-
A-003-4	A-003	Dogbone/Double-pin	No	1723.7	-
A-003(90deg)-1	A-003 (90 deg. Direction)	Dogbone/Hydraulic	No	1555.3	-
A-003(90deg)-2	A-003 (90 deg. Direction)	Dogbone/Hydraulic	No	1608.8	-
A-003(90deg)-3	A-003 (90 deg. Direction)	Dogbone/Hydraulic	No	1562.0	-
A-004-30	A-004	Dogbone/Double-pin	Yes	974.0	0.0197
A-004-1	A-004	Strap/Double-pin	No	937.4	-
A-004-2	A-004	Strap/Double-pin	No	894.2	-
A-004-3	A-004	Strap/Double-pin	No	978.5	-
A-004-5	A-004	Dogbone/Double-pin	No	974.4	-

Table 2. Test Result Summary Table – continued.

A-004(90deg)-1	A-004 (90 deg. Direction)	Dogbone/Hydraulic	No	736.3	-
A-004(90deg)-2	A-004 (90 deg. Direction)	Dogbone/Hydraulic	No	734.6	-
A-004(90deg)-3	A-004 (90 deg. Direction)	Dogbone/Hydraulic	No	705.6	-
A-005-2	A-005	Strap/Double-pin	No	2758.4	-
A-005-3	A-005	Strap/Double-pin	No	2652.6	-
A-005-4	A-005	Strap/Double-pin	No	2626.6	-
A-005-5	A-005	Dogbone/Double-pin	No	2829.0	-
A-005-8	A-005	Dogbone/Double-pin	No	2613.5	-
A-005-9	A-005	Dogbone/Double-pin	No	2721.9	-
A-005-17	A-005	Dogbone/Double-pin	Yes	2824.7	0.0151
A-005-26	A-005	Dogbone/Double-pin	Yes	2608.0	0.0159
A-005-27	A-005	Dogbone/Double-pin	Yes	2718.2	0.0163
B-1005-3	B	Dogbone/Hydraulic	No	1264.0	-
B-1006-3	B	Dogbone/Hydraulic	No	1235.0	-
B-1006-6	B	Dogbone/Hydraulic	No	1218.0	-
B-1006-4	B	Dogbone/Hydraulic	Yes	1298.0	-
B-1005-4	B	Dogbone/Hydraulic	Yes	1447.0	-
B-1005-5	B	Dogbone/Hydraulic	Yes	1290.0	-

The strain at failure in table 2 is the strain measured using the virtual displacement gauge approach in the VIC-3D software where the gauge is placed in the center of and spans nearly the full length of the gauge section of the dogbone.

Some observations to note from these results are:

1. For the same laminate layup, there is little difference between results using double-pin grips and hydraulic grips – primarily because the tests that failed at the grips were eliminated from table 2
2. Though laminate A-001 is essentially two layers of laminate A-002, the failure loads are slightly less than double (about 9% less)
3. Though laminates A-003 and A-004 have twice the number of 0-degree fibers as 90-degree fibers, the strength in the 90-degree direction is much higher than would be expected from this ratio. On the average, the 90-degree strength for A-003 is only 12% lower than the 0-degree strength, and the 90-degree strength for A-004 is only 24% lower than the 0-degree strength.
4. The failure strain for laminates A-001 through A-004 are in the 0.017 to 0.021 range, while the A-005 laminate is lower and between 0.015 to 0.016
5. The quasi-isotropic laminates of material B have about 40% higher strength than the thicker quasi-isotropic laminate A-001

Photos of failed specimens from laminates A-001 through A-005 are shown in figure 6. The speckled white and black patterns on the specimens are from paint that was applied to allow for position tracking used in the full field measurements. The actual specimen colors are as shown in figure 3. The failure characteristics are similar for laminates A-001 through A-004. The damage region localized at the failure location shows fiber fracture and pull out, and local delamination. Samples of laminate A-005 have significantly different characteristics with larger regions of delamination and longer fiber pullout lengths. For some of specimens of laminates A-001 through A-004, there is also noticeable longitudinal splitting starting from the free edge of the 1-inch-wide gauge section that propagates up into the dogbone region. Specimens of laminate A-005 have this edge splitting characteristic as well as noticeable longitudinal splits that initiate at the fracture location. No photos from material B laminate tests are available. But as will be described in a subsequent section, failure initiated fairly early in the loading profile and the specimens developed large delaminations prior to failure. So even though the load carrying capability of the material B laminate was large, the structural response was not considered acceptable.

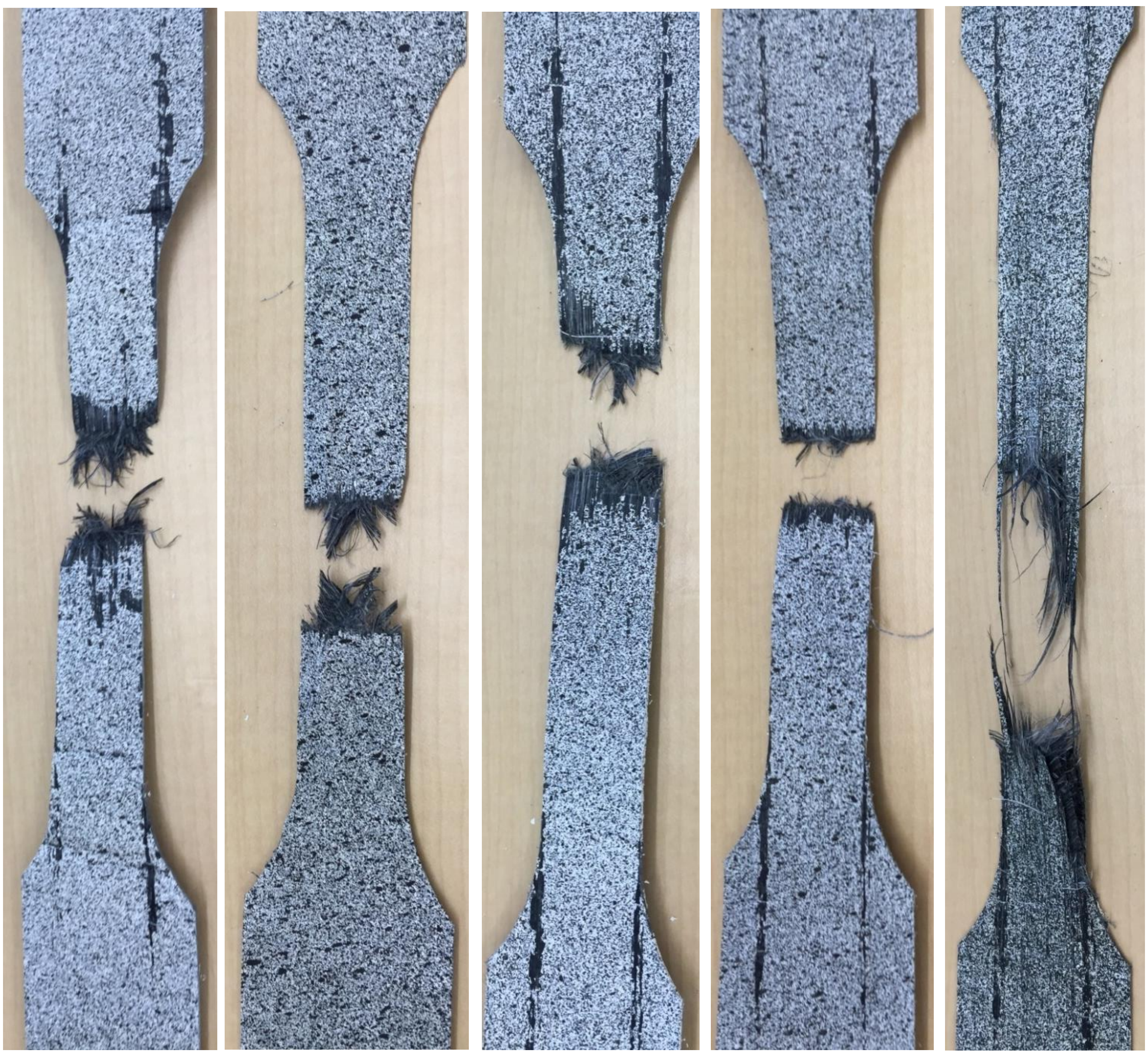

Figure 6- Photos showing typical failures for samples of laminates A-001, A-002, A-003, A-004, and A-005 (from left to right).

Full-Field Results – VIC-3D measurements were used to determine the load – strain response and strain uniformity of the flexible composite. (Note the speckled white and black patterns on the specimens in figures 5 and 6 were applied to allow for position tracking used in the full field measurements.) Figures 7, 8, and 9 show the load-strain results and the Poisson's ratio results using the virtual extensometer capability of the VIC-3D software for samples of laminates A-001 through A-004 (figure 7), A-005 (figure 8), and B (figure 9). Axial strain in these figures is determined from surface position tracking of a path extending approximately the full length down the middle of the 1-inch-wide test section between the dogbone transitions in the samples. The axial strain is then computed as normalized length change Δl / l for this path. Transverse strain is similarly computed as the normalized length change for a path perpendicular to the load direction, and Poisson's ratio is calculated as the negative of the ratio of these two quantities.

If failure strain is defined as strain at maximum load capability, the left plot in figure 7 shows that the load-strain (and thus the stress-strain) response of laminates A-001 through A-004 is extremely linear all the way up to failure. The failure strain ranges from a low of 0.017 (for A-003-29) to a high of 0.021 (for A-002-20) for these specimens. The corresponding Poisson's ratio is plotted for these samples in the right plot of figure 7. According to lamination theory, the Poisson's ratio for these laminates would be constant for each sample, from a low of about 0.21 (for A-003) to a high of about 0.35 (for A-001 and A-002). The Poisson's ratio in figure 7 is load dependent and typically starts high at low load, and then rapidly reduces to a minimum as the load is increased. As load is further increased it increases again but at a much slower rate than the initial change. The cause for this behavior is unknown at this time, however, along with the load-strain response, it appears to be useful as an indicator for changes in internal loadpath such as those caused by internal damage.

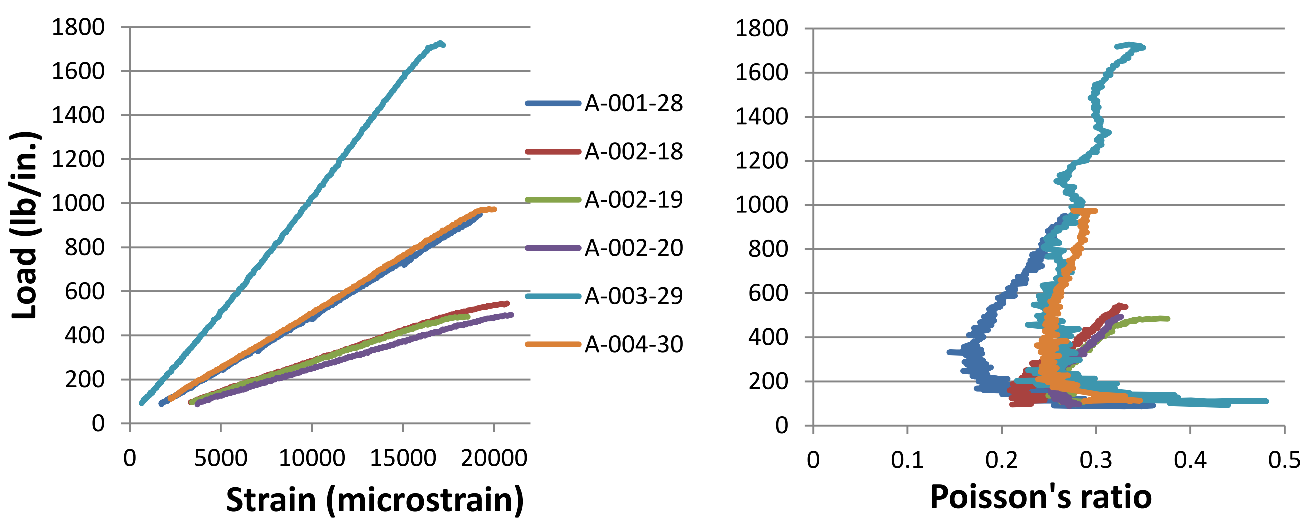

Figure 7 – Strain and Poisson's ratio from VIC-3D virtual extensometer for samples of laminates A-001 through A-004.

The load-strain and Poisson's ratio response for samples of laminate A-005 are shown in figure 8. This 0-degree-fiber-dominated laminate has a fairly linear load-strain response, but not as linear as laminates A-001 through A-004. The failure strains are also a bit lower than A-001 through A-004 with a low value of 0.0151 and a high value of 0.0163. The lamination-theory Poisson's ratio for A-005 is 0.77, which is much larger than the A-001 through A-004 laminates which have a higher percentage of cross-plies. The A-005-27 sample has the highest degree of nonlinearity, and also qualitatively different Poisson's ratio response than the other A-005 samples.

The load-strain and Poisson's ratio response for the three material B laminate samples are shown in figure 9. There is a discrete change in slope in the load-displacement plots at the 200 lb load level. This is also the load level at which there is an inflection point in the Poisson's ratio, and above this load level the Poisson's ratio increases significantly higher than the ~0.33 predicted by lamination theory. As the load increased during these tests, delaminations of the surface plies were observed, which increased lateral strain measurements that are reflected in the high Poisson's ratio. However, the laminate retained the ability to carry a load level that was ultimately much higher than the thicker A-001 laminates – a result of the high volume fraction of the stronger and stiffer Dyneema fibers in the B laminates.

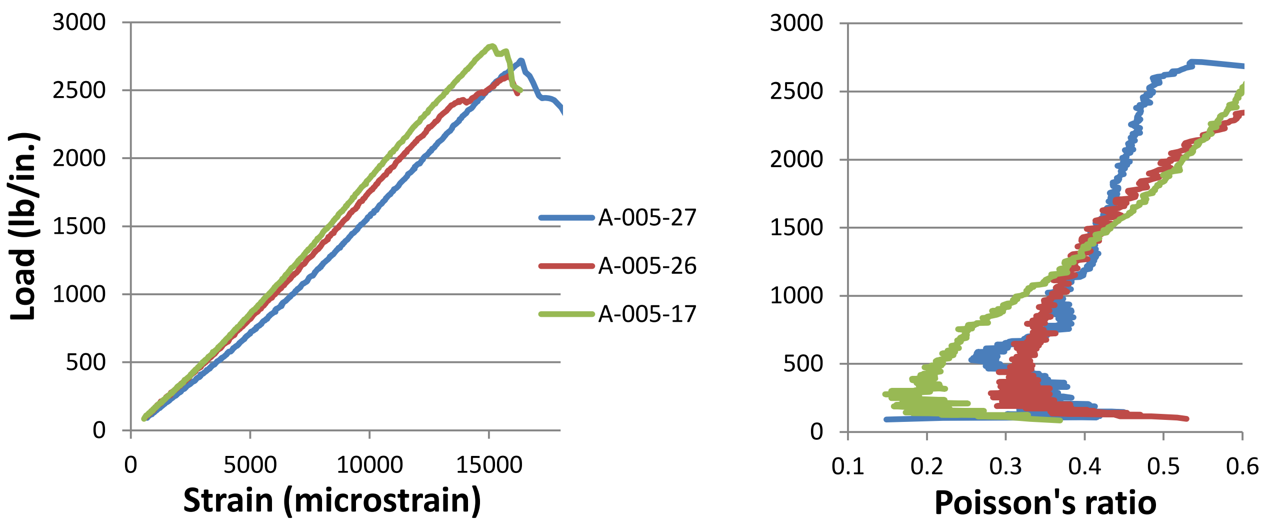

Figure 8 – Strain and Poisson's ratio from VIC-3D virtual extensometer for samples of laminate A-005.

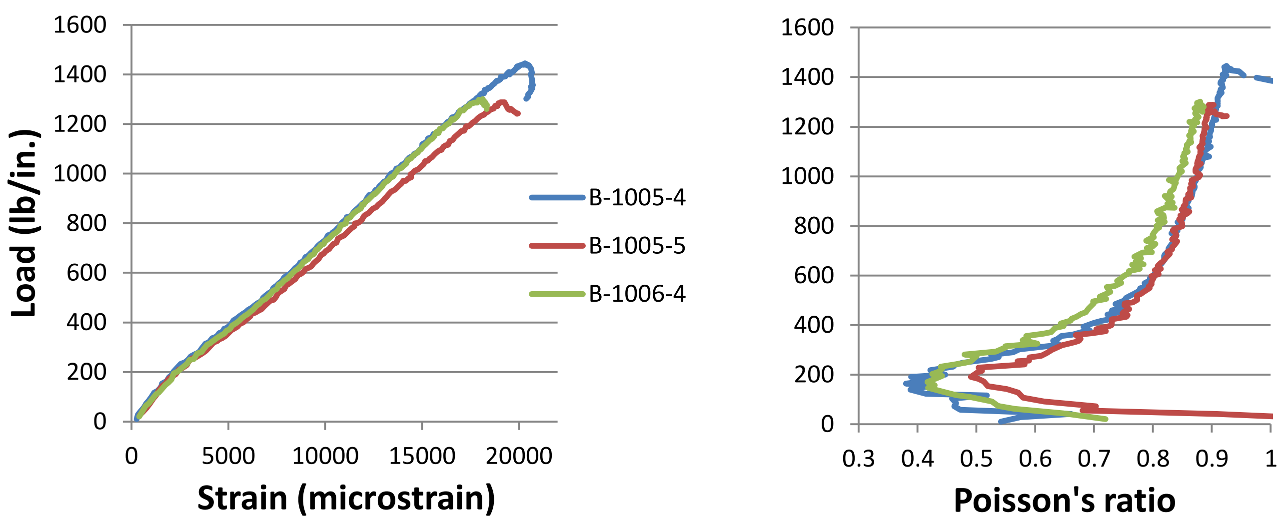

Figure 9 – Strain and Poisson's ratio from VIC-3D virtual extensometer for samples of laminate B.

Figures 10 through 14 show full-field strain measurements of axial strain for laminates A-001 through A-005, respectively. These measurements are displayed for 4 load levels: 25%, 50%, 75% and 90% of the peak load for each individual sample. Notice that a different full scale strain value is shown in the legend for each load level. No attempt was made to prepare test specimens without creases (which are caused by handling of the laminates), and the speckled paint needed for VIC-3D measurements was applied over any creases that were in the specimens. Thus, areas with creases will flatten out under load and show up as having high localized strains in the full field data. And for some samples (e.g., in the center of the dogbone section in figure 10), no strain is measured at a crease location because the high uncertainty in resolving the speckle pattern in the vicinity of this discontinuity was beyond the limits defined in the post-processing software. However, the strain measurements are otherwise generally uniform over the test section, and there is a significant reduction in the effectiveness of the dogbone ends to transfer load as the laminate layups go from quasi-isotropic (figures 10 and 11) to moderately tailored (figures 12 and 13) to highly oriented (figure 14). Similar data are shown in figure 15 for sample B-1005-4 and it shows a high level of strain uniformity at all load levels.

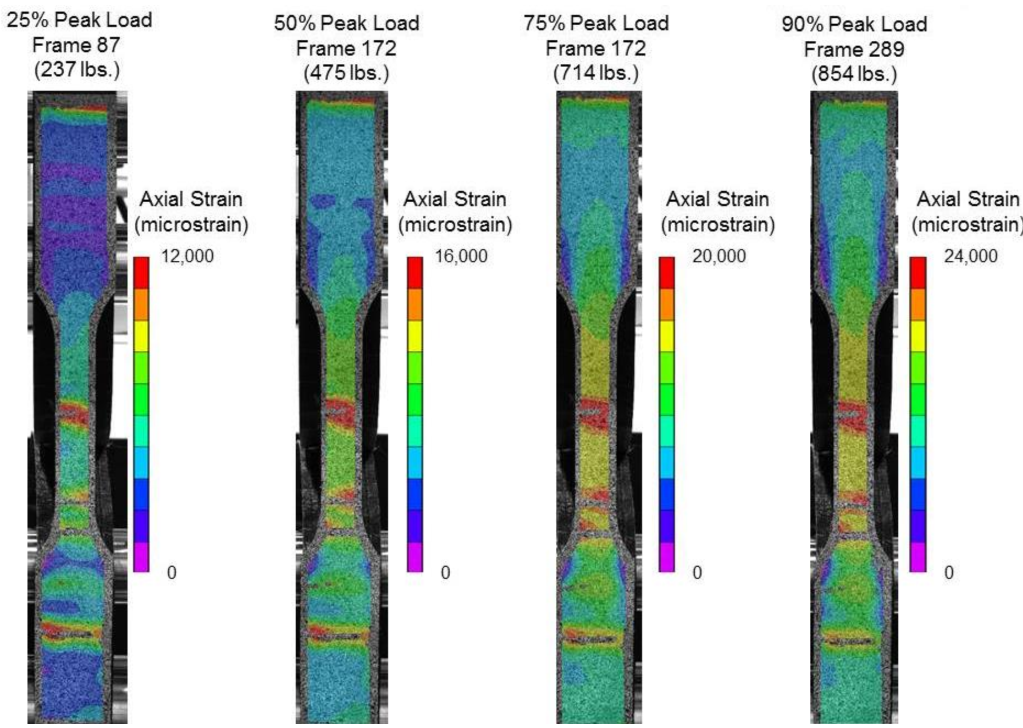

Figure 10 – Full field measurement of axial strain at 4 load levels for sample A-001-28.

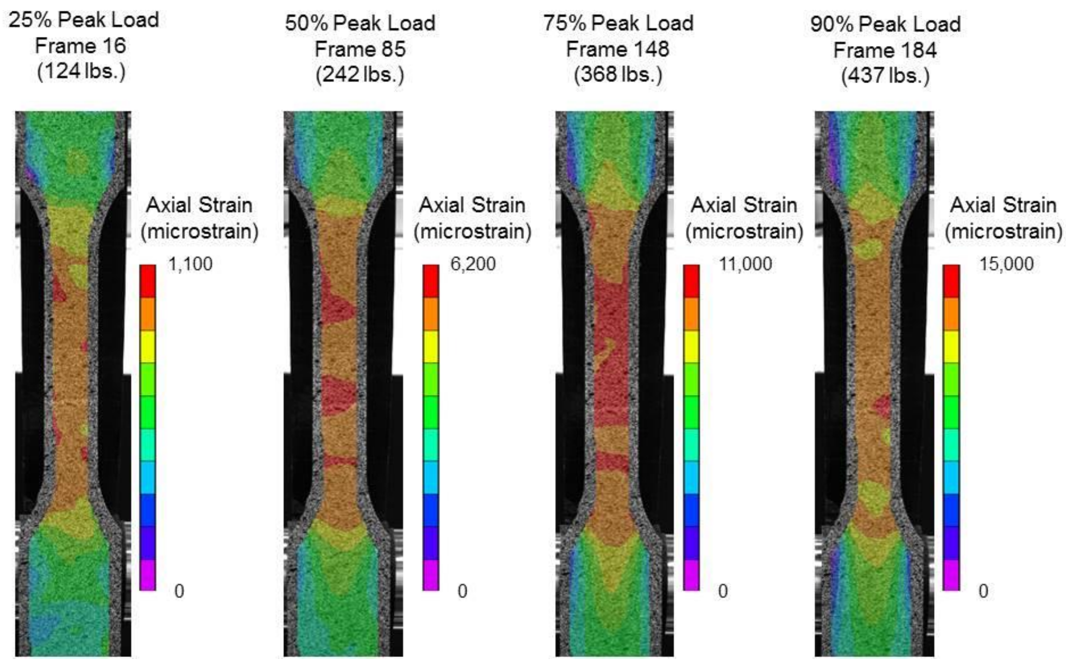

Figure 11 – Full field measurement of axial strain at 4 load levels for sample A-002-19.

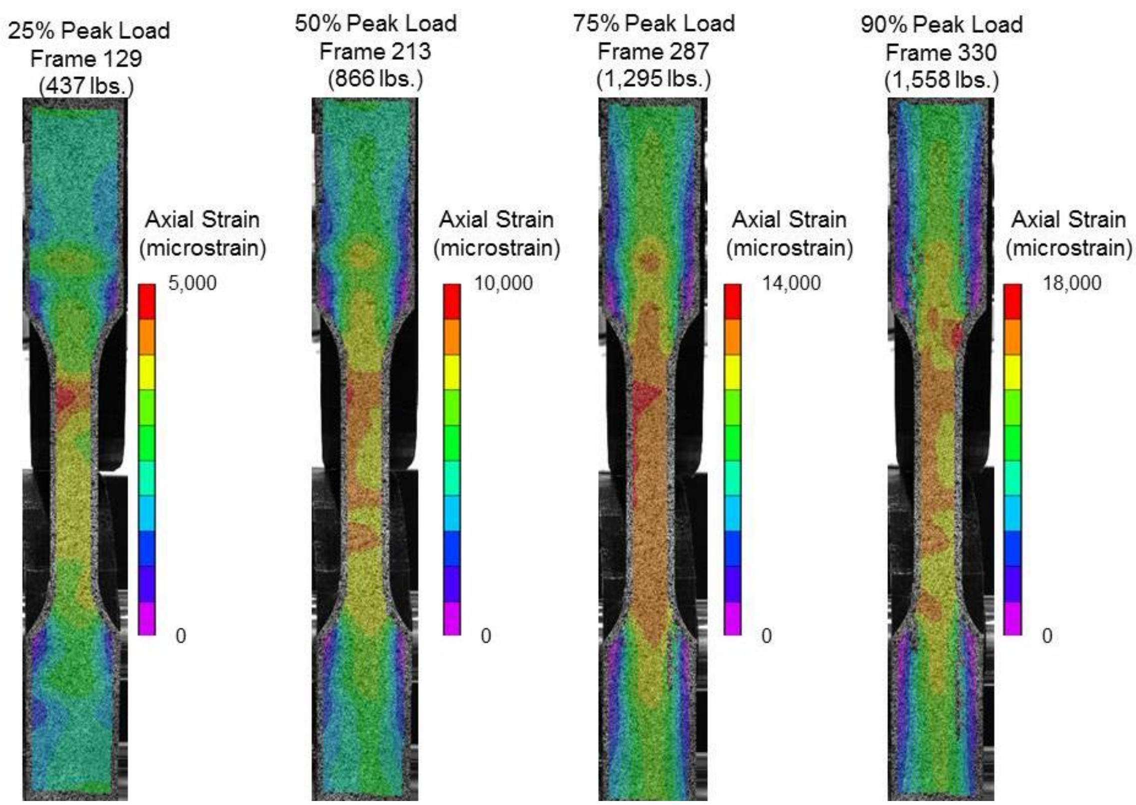

Figure 12 – Full field measurement of axial strain at 4 load levels for sample A-003-29.

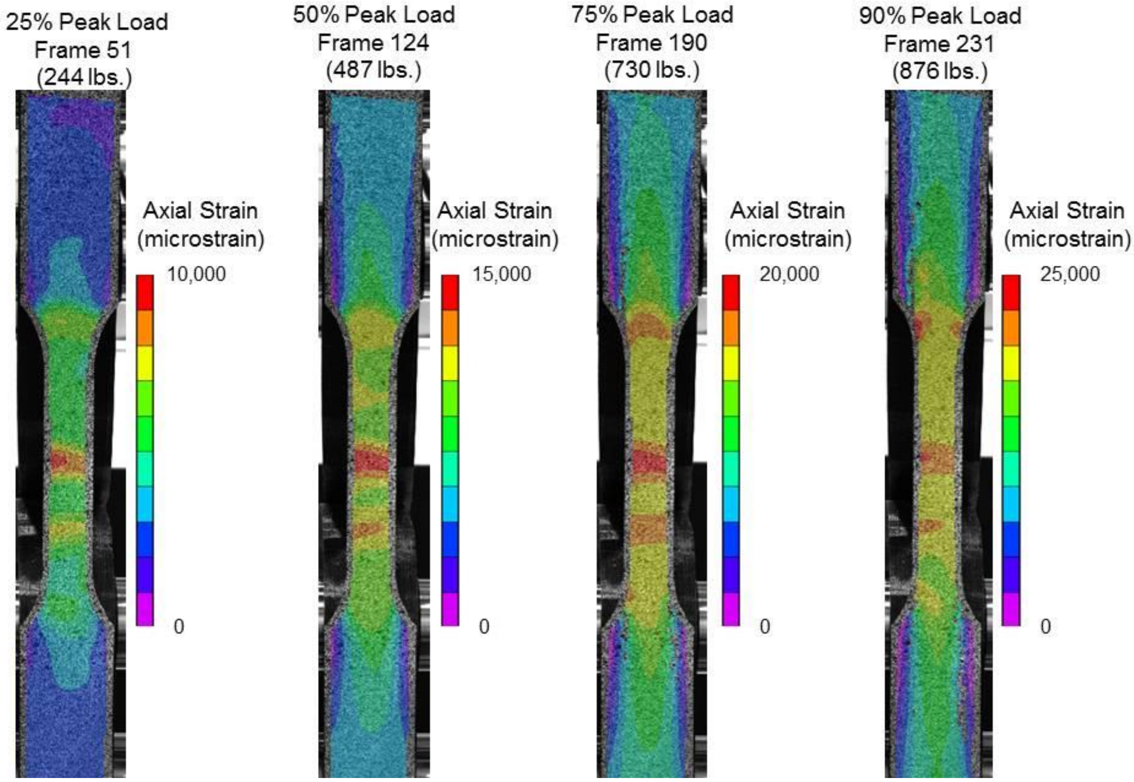

Figure 13 – Full field measurement of axial strain at 4 load levels for sample A-004-30.

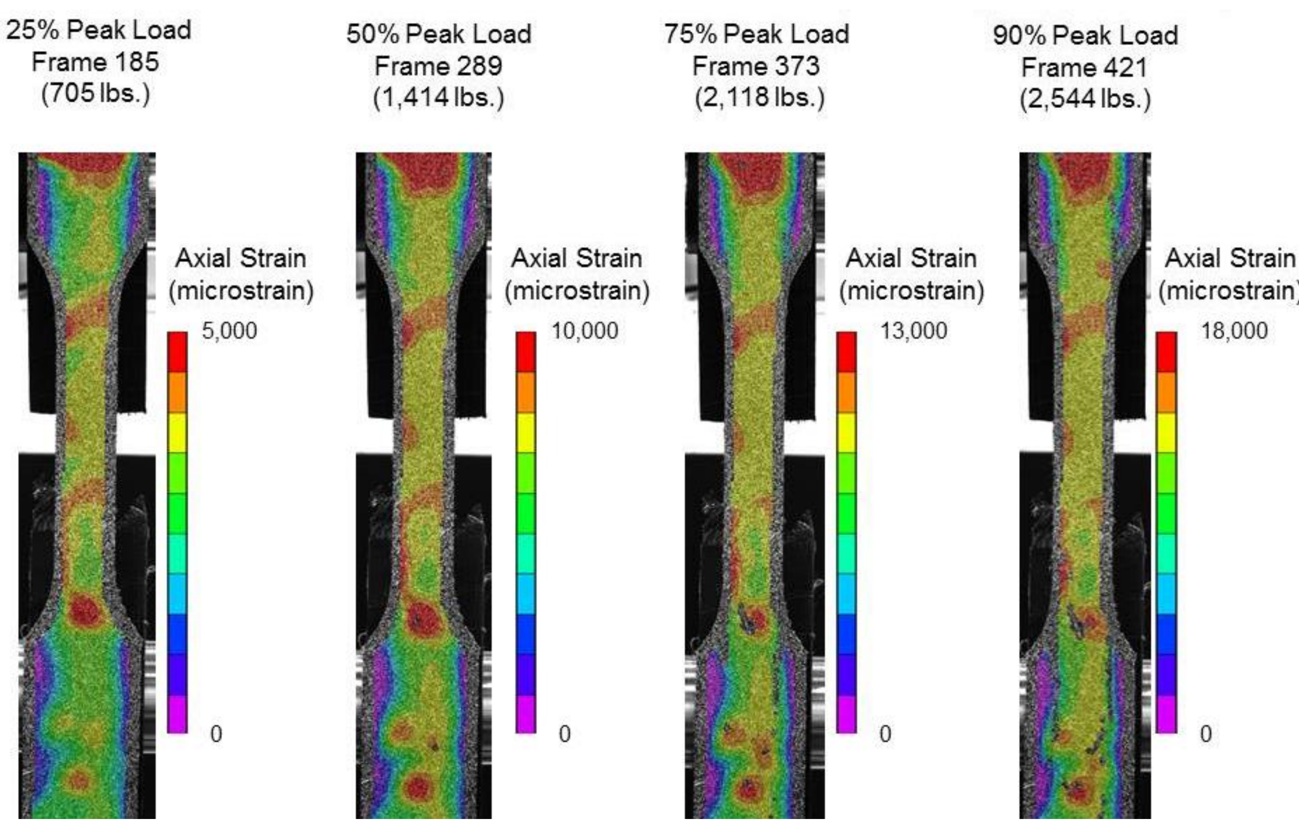

Figure 14 – Full field measurement of axial strain at 4 load levels for sample A-005-17.

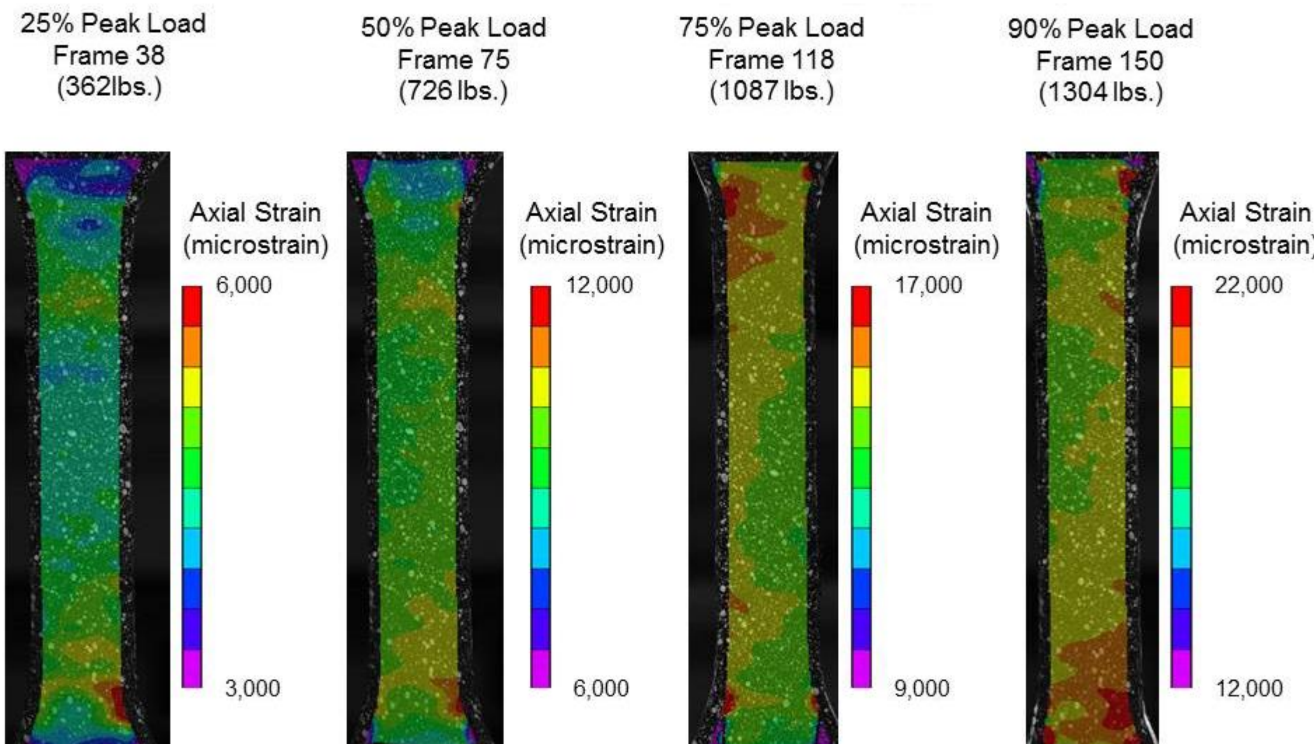

Figure 15 – Full field measurement of axial strain at 4 load levels for sample B-1005-4.

Estimation Methodology for Lamina/Laminate Design Properties

The data described in the preceding sections indicate that flexible laminates, especially the materials used for the A laminates, have some attractive characteristics. The laminated material has a high level of flexibility, and when creases form due to folding, the effect on failure loads is small. It has a linear load-strain response compared with woven material, for example, compare figures 7 and 8 with figure 2; and it has failure strains approaching 2%. However, the present exploratory testing effort is not sufficient to develop a design database, so the following methodology was used to estimate design properties:

<u>Determination of Lamina Properties</u> – To determine stiffnesses of arbitrary laminate layups of material A, the following process is used to determine the lamina properties E_{11}, E_{22} , G_{12} , and ν_{21}.

1. Assume a uniaxial lamina of material A has direction 1 in the fiber direction, direction 2 in the transverse direction, and that typical polyester matrix properties can be used for the transverse stiffness and Poisson's ratio
    a. Young's module, E_{22} , = 150000 psi
    b. Poisson's ratio, ν_{21} , = 0.4
    c. Assume, $G_{12} = E_{22} / 2(1 + \nu_{21})$ (e.g., ref 10)
2. Utilize VIC-3D test data from laminates A-001 through A-004 to calculate laminate axial stiffness E_{xx} , these data are shown in table 3 (note: to obtain stress from load, the idealized sample thickness obtained from the average ply thickness for all laminates times the number of layers shown in table 1 was used rather than thickness measured for each sample).
3. The only unknown lamina property is Young's modulus, E_{11} . Use least squares fit to E_{xx} data in table 3 and lamination theory (e.g., ref. 11) for the laminates in this table to estimate E_{11}

Using the data in table 3, the stiffness for E_{11} was determined to be 4094000 psi. The estimation was repeated for doubling the stiffness E_{22} in 1) above and it only reduced E_1 by 3%. Similarly, using half the value of ν_{21} in 1) above, the estimation of the stiffness for E_{11} changed only in the fifth significant figure. So the above lamina property determination methodology appears to be fairly robust.

Table 3 – Longitudinal Young's modulus from VIC-3D test data, and value computed using estimated lamina properties.

Sample Number	E_{xx} - VIC-3D (psi)	E_{xx} - fit (psi)	Failure Strain - VIC-3D	Failure Strain - est
A-001-28	1248000	1421000	0.0192	0.0199
A-002-18	1418000	1421000	0.0207	0.0201
A-002-19	1428000	1421000	0.0184	0.0178
A-002-20	1295000	1421000	0.0210	0.0199
A-003-29	2347000	2348000	0.0170	0.0176
A-004-30	2310000	2115000	0.0197	0.0196

Determination of Failure Allowables – Considering only samples of laminates A-001 through A-004, only 6 failure strains were directly obtained because only 6 tests from this series used VIC-3D measurements. However, the linearity of the load-strain results shown by figure 7 allows for conversion of failure loads in table 2 to failure strains using the modulus data in table 3. Using the VIC-3D data for E_{xx} and obtaining sample stress by dividing the failure loads in table 2 by the idealized sample thickness (see step 2 above), an estimated failure strain can be obtained for every axial test of A-001 through A-004 laminates in that table. The good agreement of the estimated strain using this approach with the VIC-3D measured strain is shown by the rightmost two columns in table 3. Using this approach provides 26 measurements that can be used to compute a pseudo B-Basis strain allowable for material A. The mean strain for these data is 0.0188, and the standard deviation of the data is 0.0010. Assuming a normal distribution, the one-sided B-basis tolerance factors, k_B, using 26 data points from reference 12 is 1.825, and the B-Basis strain allowable is ε_{ult}= 0.0169. This allowable will be used subsequently for evaluation of the effectiveness of flexible composites compared to the state-of-the-art woven structures.

Application of Flexible Composite to In-Space Habitation Structure

A simple trade study to determine the value of a flexible composite for NASA human space exploration applications is described in this section. The application chosen for this study was a comparison of the heritage structural softgoods approach using woven straps and a structural bladder with a flexible composite approach for the cylindrical wall of a habitation structure, the Mars TransHab. The TransHab (ref. 1) consists of a lightweight graphite-composite core, surrounded by a 27-foot diameter inflatable shell and it was originally envisioned to be the habitation module of an interplanetary transit vehicle. For the present comparison, a section of the 27-foot diameter cylindrical acreage region of the inflatable shell will be sized for static strength only, and the heritage woven soft goods approach will be compared to the flexible composites investigated in the present study. For this comparison, only the pressurization loads will be utilized where a mean operating pressure of 14.7 psi will be assumed. This pressurization of the TransHab cylindrical shell induces internal loads of N_x = 2380 lb/in. in the hoop direction, and N_y = 1190 lb/in. in the axial direction. These operating pressure loads will have an appropriate factor of safety applied to establish the design loads. Adhering to NASA Standard 5001B (ref. 6), a factor of safety of 4 for softgoods structures is used while the predictable response of the flexible composite is assumed adequate to allow a factor of safety of 2 as specified for habitation modules in the standard.

Heritage woven softgoods sizing – The heritage woven softgoods design consists of a lattice of woven straps to provide the structural restraint, and a flexible bladder to provide for hermeticity. The loading is assumed to be purely biaxial, aligned with the strap lattice, and no stiffening for shear loads is specified. Vectran straps as specified in Mil-T-87130 (ref. 13) that are nominally 1 in. wide are used in this sizing. The straps assumed for this sizing are 12.5K straps for the hoop direction having a minimum ultimate tensile strength of 12,500 lb, and a mass per unit length of 0.00283 lb_m/in., and 6K straps for the axial direction having a minimum ultimate tensile strength of 6000 lb, and a mass per unit length of 0.00156 lb_m/in. (note: the unit length mass data are from unpublished measurements for materials tested and reported in reference 4). For the hoop direction, the stress resultant N_x from the mean operating pressure multiplied by a factor of safety of 4 yields N_{x-ult}. To provide sufficient strength for this internal load, the straps are spaced by w_{x-gap} as described in this equation.

$$N_{x-ult} = 12{,}500 \text{ lb} /(1 \text{ in.} + w_{x-gap})$$

And the mass per unit area for straps in the x-direction, m_x , is

$$m_x = 0.00283 \text{ lb}_m/\text{in.} /(1 \text{ in.} + w_{x-gap})$$

Using the operating pressure internal loads and the design factor of safety to establish the value of N_{x-ult} (i.e,., N_{x-ult}= 9520 lb/in.), w_{x-gap} is found to be 0.31 in., and m_x is 0.00230 lb_m/in^2. Similarly, the spacing between straps, w_{y-gap}, and the mass per unit area for straps in the y-direction, m_y , can be obtained from N_{y-ult} (i.e,., N_{y-ult}= 4760 lb/in.), and the values are 0.23 in. and 0.00124 lb_m/in^2 , respectively.
The structural shell unit mass is the sum of these two values. Note that the strength degradation of stitching together straps at their overlaps, and the additional mass which would be added from this stitching are ignored in this simple sizing exercise. The hermeticity of the habitat and the localized structural function of reacting the pressure loads in the gaps between the straps is provided by a "structural" bladder that is internal to the lattice. A single layer of coated Vectran 200-D with a unit mass of 0.00046 lb_m/in^2 is assumed for this bladder where the unit mass is derived from data in reference 14. Summing up the unit mass of the bladder with the unit masses from the straps in both directions yields a system unit mass of 0.00400 lb_m/in^2.

Flexible composite sizing – The flexible composite sizing is based on the properties obtained from the material A laminates described above. A composite allows for significant tailoring to minimize mass while reacting the loading without failure. For the present comparison, the laminate is tailored to react the biaxial loads with only a small number of plies providing shear stiffness. This is believed to be a fair assumption since there was no shear stiffness requirement used in the heritage softgoods sizing. The strain allowable, ε_{ult}, derived in the previous section is utilized in the strength evaluation. A factor of safety of 2 is used to define the ultimate load levels, which results in N_{x-ult}= 4760 lb/in. and N_{y-ult} = 2380 lb/in. Using the lamina properties for E_{11}, E_{22}, G_{12}, and ν_{21} described previously for material A, lamination theory is used to compute the in-plane stiffness matrix, $[A]$, and a laminate that satisfies the following relations will be an acceptable design

$$\begin{pmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{pmatrix} = [A]^{-1} \begin{pmatrix} N_{x-ult} \\ N_{y-ult} \\ N_{xy} = 0 \end{pmatrix}$$

and

$$\varepsilon_x , \varepsilon_y < \varepsilon_{ult}$$

where shear strain is neglected assuming the positive and negative 45 degree lamina are balanced. The laminate with the following stackup was found to satisfy these relations $[45/-45/(0/0/90)_{12}/0/0/0.5*90]_s$. This 81 ply laminate has only single stacks of ±45° plies on the external surfaces and approximately a 2:1 ratio of hoop to axial plies with a total thickness of about 0.097 in. Using the average composite density of 0.0309 lb_m/in^3, the system unit mass for the flexible composite is 0.00300 lb_m/in^2. This system unit mass for the flexible composite is a 25% mass saving over the heritage softgoods system unit mass.

Qualitative Evaluation of Flexibility and Handling – The sized laminate of material A for the TransHab application is significantly thicker than any of the laminates tested in this study. Of the tested laminates, A-003 was the thickest of the crossplied laminates (i.e., A-001 through A-004) at 0.037 in., but the laminate sized for TransHab was 0.097 in. - over twice as thick as any tested. A qualitative evaluation of the flexibility and handling characteristics of a laminate approximately simulating the thickness of this TransHab laminate was performed using the laminates on hand. One layer of A-004 material was sandwiched between two layers of A-003, and they were bonded together using a room-temperature-curing, liquid-rubber adhesive. This laminate is slightly thicker than 0.1 in., and it is pictured in the left panel of figure 16. Excess adhesive can be seen on the edges of this laminate. This sample was used to evaluate laminate flexibility and effects from handling. The laminate was found to be noticeably stiffer than those described previously. However, it was easily bent, and the upper-right panel of figure 16 shows the laminate wrapped around a 5/8 in. diameter plastic pipe. This handling resulted in the formation of a number of ridges in the section of the laminate that was deformed to high curvature as shown in the lower-right panel of figure 16. These ridges are more pronounced than the creases noted in figure 4 for thinner laminates of material A. It is not known if this effect is intrinsic to the material at this thickness or if it was caused by local failures of the rubber adhesive used in creating this sample. Like the sample in figure 4, this thicker sample relaxed to be nearly flat after this handling test.

Discussion

The composite material used to produce material A laminates, though not an aerospace material (e.g., having tight material specifications for fiber waviness, tow gaps, etc.), has an attractive balance of properties. It should be noted that it was developed for use in sails of racing sailboats, and has not been optimized for NASA uses. In spite of these limitations, it has the extremely attractive features of being reasonably flexible, having a linear stress-strain response, repeatable failure modes for blocks as large as 3 co-directional plies, and relatively low scatter of failure strain. Also, the failure strains do not appear to be significantly affected by the presence of creases in the material. These qualities support the assumption that a factor of safety of 2 can be utilized in their design. Much of the experience base for composites that has been developed over many years of research and development can be leveraged to develop flexible composite structures. Some examples of this experience are: optimal tailoring of laminate layups for mass, stiffness, etc.; understanding of the mechanics of deformation and failure; fabrication technology that allows for manufacturing to precise geometric specifications including tooling and automated tape layup; non-destructive inspection; and many others. However, these conclusions are based on a very limited set of static testing. There are many unknowns that still require investigation, the most significant being the damage tolerance (including damage from material creasing, notched properties, etc.), material long-term durability, creep of the material under continual loading, additional evaluations of flexibility for thicker laminates, and leakage when used for pressure containment – especially in the presence of creases. However, the 25% mass saving estimated for the TransHab application provides impetus to take the steps to address these unknowns in future studies.

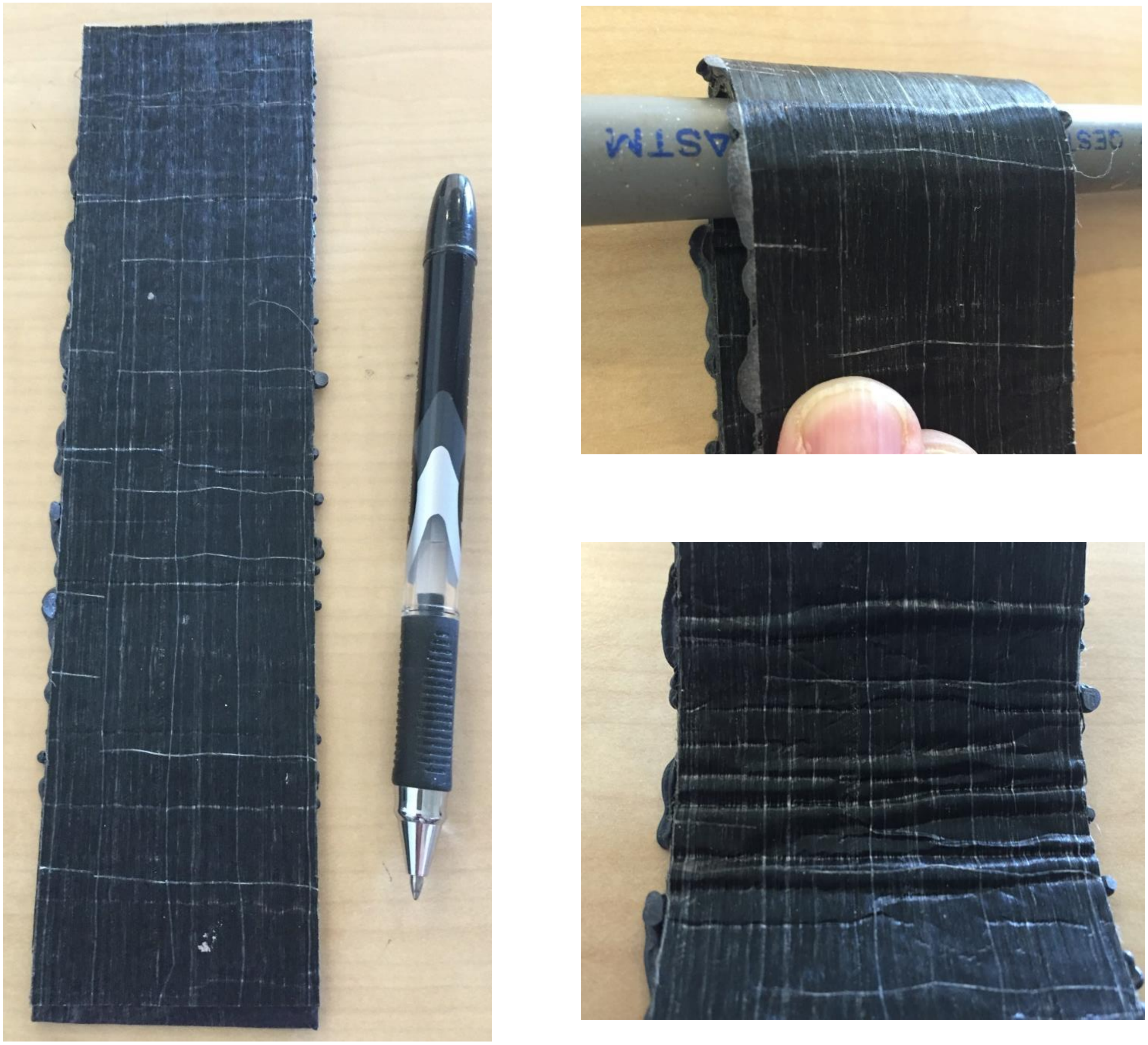

Figure 16 – Simulation of a thick laminate of material A. Left – Three layers of Material A laminates (i.e., 2 layers A-003, 1 layer A-004) adhesively bonded together prior to handling test. Upper-right – Laminate bent to conform to a 5/8 in. diameter cylinder. Lower-right – Laminate showing ridges that resulted from handling test.

Summary:

Two flexible composite material systems - identified as materials A and B herein - were investigated to determine their applicability to replace heritage woven softgoods for NASA applications. Material A is typically used for fabricating sails of racing sailboats, and material B is used for ballistic protection. The laminates studied had layups ranging from quasi-isotropic to highly aligned in one direction in order to provide data over a range of laminates that would be of interest for space applications. General material characteristics such as effects due to handling and physical properties were obtained. Mechanical properties – stress- strain response, static strength, and failure characteristics - were also obtained from uniaxial testing and full field strain measurements. The results showed material A had an attractive balance of properties – linear response nearly to failure and repeatable failure modes having low scatter in failure strains - while material B showed initial failures early in the mechanical loading. The test data for material A were analyzed to develop lamina elastic constants and a failure criterion that could be used for design. These data were subsequently used to compare flexible composites to heritage woven softgoods for a NASA habitation pressure vessel application. Using the same static strength (and a hermeticity) requirement, the flexible composite design was estimated to save 25% over the woven softgoods design. Though there are many unknowns that were not addressed in the present study, the flexible composite material merits further study.

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14. ABSTRACT
Payload volume for launch vehicles is a critical constraint that impacts spacecraft design. Deployment mechanisms, such as those used for solar arrays and antennas, are approaches that have successfully accommodated this constraint, however providing pressurized volumes that can be packaged compactly at launch and expanded in space is still a challenge. One approach that has been under development for many years is to utilize softgoods – woven fabric for straps, cloth, and with appropriate coatings, bladders – to provide this expandable pressure vessel capability. The mechanics of woven structure is complicated by a response that is nonlinear and often nonrepeatable due to the discrete nature of the woven fiber architecture. This complexity reduces engineering confidence to reliably design and certify these structures, which increases costs due to increased requirements for system testing. The present study explores flexible composite materials systems as an alternative to the heritage softgoods approach. Samples were obtained from vendors who utilize flexible composites for non-aerospace products to determine some initial physical and mechanical properties of the materials. Uniaxial mechanical testing was performed to obtain the stress-strain response of the flexible composites and the failure behavior. A failure criterion was developed from the data, and a space habitat application was used to provide an estimate of the relative performance of flexible composites compared to the heritage softgoods approach. Initial results are promising with a 25% mass savings estimated for the flexible composite solution.

15. SUBJECT TERMS
Expandable space structures; Flexible composites; Woven structure

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