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USE OF GRAPHITE/EPOXY COMPOSITES IN SPACECRAFT STRUCTURES: A CASE STUDY

The feasibility of using graphite/epoxy composites in the fabrication of critical structures in support of APL programs is demonstrated by a redesign of the POLAR BEAR satellite's center support structure. The new technology can also be applied to other programs throughout the Laboratory and elsewhere.

INTRODUCTION

The emergence of high-performance composite materials can be attributed largely to the demands of the aerospace industry for lighter, stronger, and stiffer structures. One example is the high degree of thermal stability required by optical structures used in space applications to assure the accuracy of sensitive instruments. The use of composite materials in place of metals for critical load-bearing applications requires a marked transition in design philosophy. Although fabrication and design of structures from composite materials pose challenges, the potential rewards typically outweigh the real or imagined potential barriers.

Major users and suppliers of composite materials have developed a large body of technical and analytical expertise in composite structural design. The transfer of the technology in a usable form to designers of potential composites applications is the present challenge. The successful integration of conventional design practices and experience with unfamiliar analysis and technology requires that the designer develop some appreciation for the overall process. The first composite structure design for an organization can thus be the most challenging, regardless of its intrinsic complexity.

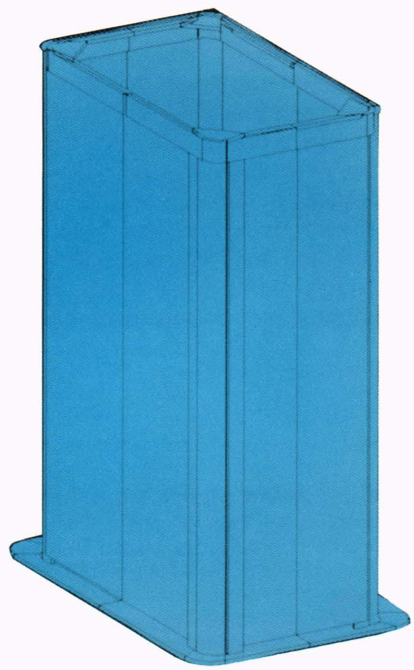


Figure 1—Composite graphite/epoxy center support structure for the POLAR BEAR satellite.

BACKGROUND

The process described is one by which a satellite substructure, originally designed at APL, was redesigned to be fabricated from a graphite/epoxy fiber-reinforced composite material. Specifically, the structural component is the center support column of the OSCAR, TRANSIT, HILAT, and now the POLAR BEAR satellites. Figure 1 is a graphic representation of the component. The basic OSCAR concept was used in all the spacecraft, with modifications made as required by the particular mission. Originally, the component was specified to be constructed of fiberglass (OSCAR) and was then upgraded to a titanium alloy (TRANSIT, HILAT, and POLAR BEAR). The composite structure designed for POLAR BEAR and described here was not used in the flight structure, primarily because of schedule constraints.

The design chosen for the case study was intentionally simple. The case study was intended to provide an example of the process of applying existing composite design and analysis methodology to satisfy the same design constraints as those of the original structure.

The structural component chosen for the case study supports instruments carried on the satellites mentioned above, two of which were designed and built at APL. Instrument packages are carried either on a platform supported by the center column or hung on boards that extend fin-like from the corners of the column. The structure, shown schematically in Fig. 2, is loaded both statically and inertially, primarily during launch, with axial, shearing, and moment loads about both axes of its base plane. In orbit, the structure serves primarily to remove heat generated by the experiment packages and withstand thermal loads imposed by differential solar heating.

The design of the flight support structure of the POLAR BEAR satellite consists of a built-up assembly

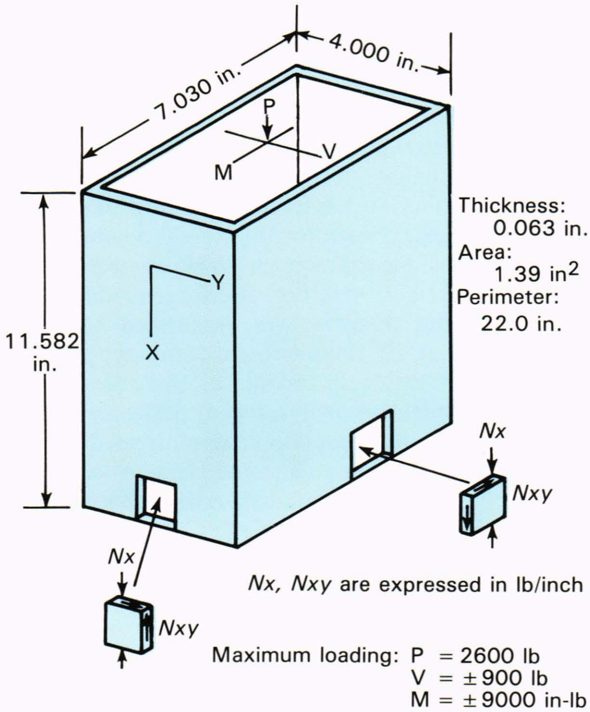


Figure 2—Schematic diagram of the loading for the center support column.

fabricated from Ti-6Al-4V alloy, a relatively common choice for space structures. The current design consists of 14 brackets and stiffeners and 184 fasteners. The unit weighs about two pounds and is fabricated entirely in APL's shops. As successive editions of the satellite were configured with more extensive and heavier instrumentation, the overall weight margin narrowed and the weight contribution of all the supporting structures became more significant.

In considering replacement of the titanium alloy material in the center support column with an advanced fiber-reinforced composite, a weight savings of about 50 percent was desired. In absolute numbers, this one-pound saving is not large; however, it was considered to be significant when achieved in such a small structure. This indicated the potential for significant weight savings in other structural components of weight-critical satellites. By using a composite material and existing technology for fabrication, the parts count could also, in principle, be reduced to one. Top and bottom brackets would be integrally fabricated with the side panels, and fasteners would be eliminated.

DESIGN CONSIDERATIONS

In transferring the design of the center support structure to composite materials, the following requirements were set:

- Strength to equal or exceed maximum design loads, plus a safety factor
- Stiffness to equal or exceed that of the existing titanium alloy structure
- Overall dimensions coordinated with other structures (the same configuration as the built-up titanium structure)
- Longitudinal thermal conductivity to equal or exceed the existing titanium alloy structure
- Buckling stability to equal or exceed the existing titanium alloy structure

A number of design concepts were considered within the overall constraint that the composite structure must be interchangeable with the titanium structure.

The first concept was a simple substitution of graphite/epoxy for titanium in the side panels. Even though that option is the simplest, it was considered unattractive because only minimal weight savings would be realized. The second concept was an integral composite column with machined-metal end pieces. That concept was deemed practical, having a greatly reduced parts count and assembly effort, but it was not considered further because it was difficult to fasten or bond the rectangular composite tube to the metal ends and there was less-than-optimum weight reduction.

The third concept that was considered was an integrally fabricated structure (a one-piece column with no fasteners). This concept was judged to be the best design. The mandrel for forming the column and the external mold, however, are considerably more complex and therefore more expensive. A potential for fiber breakage exists in the transition area from the hollow rectangular section to the end flanges. The only post-molding fabrication effort required would be drilling holes in the end flanges and facing the surfaces that mate the column to the spacecraft. The remainder of this discussion describes the design and analysis of the integral center support structure concept.

ANALYSIS

The basic analyses conducted for the design were stress, strength, stiffness, thermal, and buckling. The results of the analyses are discussed in the following paragraphs.

The quasi-static stress analysis was performed using loads and variations of those loads shown in Fig. 2. The applied loads were translated to in-plane loading of rectangular plates representing the narrow and wide sides of the support column. Each plate and each loading configuration were analyzed separately using a computer implementation of classical laminated plate theory.[1] The theory allows the analysis of plates that are constructed by laminating orthotropic layers (plies) so that the material property axes are oriented at varying angles. Deflection, curvature, and stresses in each layer are typical outputs of such analyses. Similar programs are available commercially[2] and through government-sponsored software clearinghouses.[3] As with any computer program, an understanding of the underlying theory, the meaning of the input and output variables, and the nomenclature and notation is essential for intelligent, confident use. Several good introductory texts on composite materials[4,5] develop the laminated plate theory in sufficient detail and consistency for a potential user of a related computer program to be able to compensate for any inadequacies of the program documentation.

Graphite/epoxy composites were chosen for the present study because of their superior mechanical (stiffness, strength, and fatigue) and thermal properties and their wide use in aerospace structures. Several candidate graphite/epoxy materials were considered: high tensile strength, high modulus, and ultrahigh modulus. Two types of fiber stacking sequences were considered for the analysis: Type I $[0, \pm 30]_{2s}$ and Type II $[0,90,+45,-45,90,0]_s$. Both types consist of 12 laminates with the orientation angles between adjacent laminates as shown in the brackets. Both types are symmetric as indicated by the s subscripts, Type I being doubly symmetric. Reference 6 contains a detailed explanation of accepted laminate ply numbering schemes. Typical properties of these materials are shown in Table 1. In addition, some material properties of Ti-6Al-4V alloy are provided for comparison. The table shows a typical weight savings of approximately 50 percent when graphite/epoxy is substituted for titanium. Also, for the ultrahigh modulus Type II laminate, elastic moduli exceed those of the titanium, while the strengths approach those of the titanium alloy.

For the design loads, significant strength margins existed for all the combinations of composite material. The projected failure loads, based on the Tsai-Hill failure criterion,[7] exceeded the design load in all cases, typically by a factor of 10.

Matching the stiffness requirement of the Ti-6Al-4V alloy proved to be a more constraining factor. The laminated plate theory provides effective longitudinal and transverse stiffnesses for plates of specified ply properties and orientations. Only the combination of the Type II laminate and ultrahigh modulus material yielded values of both longitudinal (x) and transverse (y) stiffnesses that exceeded the values for the titanium alloy; therefore, the P-75 (2034D) ultrahigh modulus material was used exclusively for the balance of the analysis.

The first approach to the buckling analysis was to consider buckling of the narrow and wide panels when they are subjected separately to compression and shear loads. For buckling in panels made of isotropic homogeneous materials, there exist numerous closed form or semi-empirical solutions for a variety of boundary conditions. Solutions also exist for some laminated composites having certain stacking sequences and ply orientation symmetrics. Such laminates are termed orthotropic. Nonorthotropic laminates can sometimes be treated as quasi-orthotropic if they are thick enough. Unfortunately, the stacking sequence chosen for the present design was not orthotropic and did not meet this thickness requirement; therefore, solutions were not directly available.

The buckling analysis was performed using the MSC/NASTRAN®[8] finite-element computer program. Because the program is capable of three-dimensional analysis, the buckling analyses were performed on the open-ended, flat-sided support column, treated as an integral structure. Rectangular plate elements were used and simply supported boundary constraints were imposed along the vertical edges of the box. The actual constraint condition lies between simply supported and clamped; therefore, the simply supported case was considered a lower bound for the analysis.

As with the strength analyses, several different loading cases were considered. Axial, moment, and combined loadings were imposed on the structure and critical buckling loads were calculated from the computer output for each loading case. The buckling analysis results are shown in Table 2. The margin of safety for buckling was satisfactory in all cases except one of combined axial and moment loading, where the finite-element prediction yielded a −0.05 safety margin. However, in the actual structure, horizontal stiffeners are fitted at approximately mid-height, effectively reducing the unsupported span by one-half and increasing the critical buckling load and safety margin.

DISCUSSION

The design of composite components for spacecraft is a reasonably straightforward extension of design with conventional engineering materials. A new set of design

Table 1—Typical material properties.

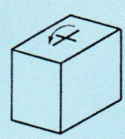
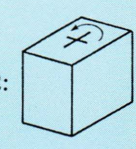
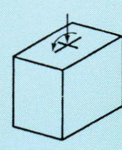
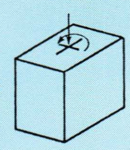
*Material Type**	*Laminate Type***	*Laminate Density* (lb/in^3)	*Moduli* ($\times 10^6$ *psi*) *Elasticity* x	y	*Rigidity*	*Strengths* ($\times 10^3$ *psi*) *Tensile* x	y	*Compressive* x	y	*Shear*
HTS	I	0.057	12.4	2.0	3.1	91.0	9.0	65.0	16.0	61.0
(T300/5208)	II	0.057	8.7	8.7	2.1	89.0	89.0	123.0	123.0	45.0
HMS	I	0.058	15.4	1.7	4.0	74.0	8.0	56.0	14.0	57.0
	II	0.058	12.3	12.3	2.9	80.0	80.0	123.0	123.0	41.0
UHMS	I	0.061	21.3	1.9	5.9	68.0	8.0	60.0	10.0	51.0
(2034-D)	II	0.061	18.7	18.7	4.2	78.0	78.0	112.0	112.0	36.0
(P-75)										
Ti-6Al-4V	—	0.160	16.5	16.5	6.1	130.0	130.0	130.0	130.0	78.0

*Material types: HTS-high tensile strength; HMS-high modulus; UHMS-ultrahigh modulus

**Laminate Type I: $[0, \pm 30]_{2s}$; Type II: $[0,90,+45,-45,90,0]_s$

Table 2—Summary of buckling analysis.

	Axial Load (lbf)			*Moment (in-lbf)*		
Loading Case	*Critical*	*Design*	*Margin of Safety*	*Critical*	*Design*	*Margin of Safety*
Axial Only	8052	2600	3.10	—	—	—
Moment Only						
Case 1	—	—	—	52,960	9000	5.88
Case 2	—	—	—	18,400	9000	1.04
Combined Loading						
Case 3	8020	2600	2.08	20,214	9000	1.25
Case 4	4921	2600	0.89	8,521	9000	-0.05

Case 1: 
Case 2: 
Case 3: 
Case 4: 

tools (laminated plate theory in this case) often is required, but the principles are similar. The experience factors often referred to as "engineering judgment" may be misleading in some cases. For instance, composite laminates may exhibit effective Poisson ratios ranging from negative values[9] to greater than unity—a considerable variation from isotropic materials, where that value is always between 0 and 0.50. Composites also offer the potential for zero or even negative coefficients of thermal expansion, which may result in large, often catastrophic, internal stresses when significant temperature excursions occur.

The material cost of the integral graphite/epoxy composite structure would be very low. Labor costs would be expected to be lower for the composite structure, considering the reduction in parts count, but fabrication of the integral structure from composite material would require autoclave facilities and technical expertise in composites fabrication. The APL Engineering and Fabrication Materials Laboratory has acquired these facilities and expertise and is currently developing in-house composites manufacturing techniques. A graphite/epoxy composite center support structure will be fabricated and tested in the future to determine its feasibility. The initial expense of tooling (a mandrel for forming the column and external molds for dimensional control) would be required for even one unit, making cost effectiveness unlikely. However, the center support column design has been used on a number of satellites and thus was an attractive candidate for a material exchange on the basis of potential cost savings. The initial expense would be amortized over the production of successive units, possibly rendering subsequent composite units less expensive than the titanium unit, which has a relatively constant cost regardless of quantity. Therefore, the cost effectiveness of composite structural components depends on the type of product as well as on the quantity of the product to be manufactured.

No article on design and fabrication using composite materials should omit some discussion of quality assurance, especially where high-reliability hardware is concerned. While composites offer many advantages over conventional engineering materials, a number of disadvantages are also present. One of the potential dangers is poor quality. Most advanced composites are fabricated by laying up prepared sheets of resin embedded with reinforcement fibers. This material, at least for thermosetting matrix materials, is tacky and each ply adheres to its neighbor as the laminate is formed. Correctness of the ply angles and stacking sequence is controlled by the competence and experience of the individual doing the work, and some care must be taken to remove air bubbles from between the plies during the process. Most laminates are cured at elevated temperatures under pressure. The laminate is enclosed in a vacuum bag when it is placed in the autoclave to ensure that all volatiles are drawn out of the autoclave and to attempt to remove any remaining air bubbles. The process is not always successful; therefore testing of the laminate is required. The volume of literature on the nondestructive evaluation of composites is large, but the most common technique for qualifying laminates is ultrasonic inspection. Without proper nondestructive evaluation, laminates should never be used in critical applications.

CONCLUSIONS

In addition to presenting a case study of graphite/epoxy composite use in a spacecraft, this study supports the following conclusions:

1. Substitution of composite for metal material is normally a straightforward process that can be accomplished with readily available design tools.

2. In single unit lots, composite hardware is not likely to be cost effective, although this is not always the case.
3. Composite hardware can be designed and fabricated in such a way that the parts count and assembly time are reduced. Since molding to near final shape is possible, machining steps are reduced.
4. Each laminate should be required to pass a nondestructive evaluation before it is used in critical applications.

REFERENCES and NOTES

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[6] DOD/NASA, *Advanced Composites Design Guide,* 1st ed., Vol. 1, *Design* (Jul 1983).

[7] The Tsai-Hill criterion is an extension of the Hencky-von Mises (distortion-energy) yield criterion for isotropic materials and is frequently used in conjunction with laminated plate theory analyses (see Ref. 5 for further details).

[8] MSC/NASTRAN is a registered trademark of the MacNeal-Schwendler Corp.

[9] C. T. Herakovich, *Composite Laminates with Negative Through-the-Thickness Poisson's Ratios,* CCMS-84-01 (VPI-E-84-10), Virginia Polytechnic Institute and State University Report (1984).

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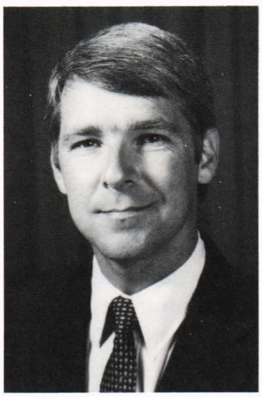
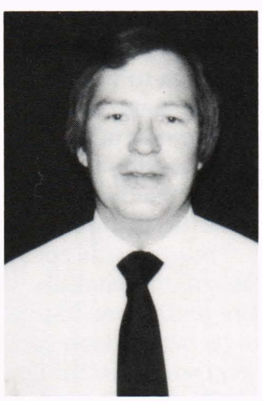
THE AUTHORS

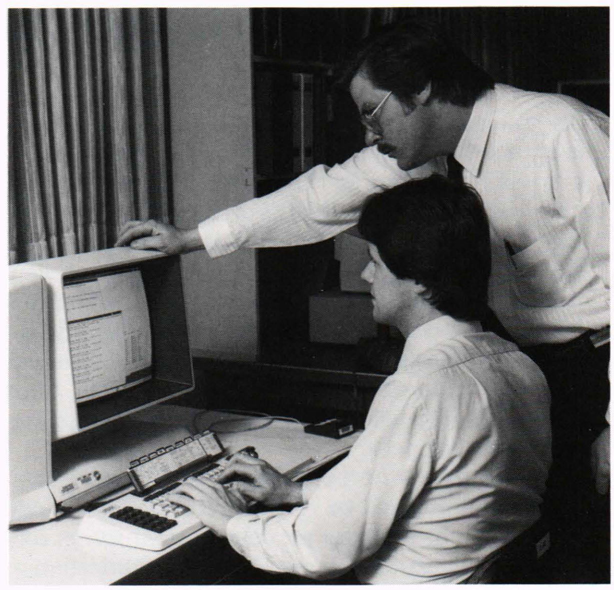


RUSSELL D. JAMISON was born in Pearisburg, Va., in 1948. Prior to receiving the Ph.D. in materials engineering science from Virginia Polytechnic Institute and State University in 1982, he worked in APL's Strategic Systems Department and was involved in Fleet ballistic missile submarine analysis. He has been a visiting senior research fellow at the University of Bath, England, where he studied damage in composite laminates using acoustic emission analysis. In 1983, he joined the Mechanical Engineering Department of the U.S. Naval Academy as an assistant professor. Since then, Dr. Jamison has been involved in the analysis of fatigue damage development and failure in composites.



O. HAYDEN GRIFFIN, Jr., was born in Abilene, Tex., in 1947. He received an M.S. degree in mechanical engineering from Texas Tech University (1971) and the Ph.D. degree in engineering mechanics from Virginia Polytechnic Institute and State University (1980). His work included research and development in finite-element analysis of composites at the Naval Surface Weapons Center, B. F. Goodrich Co., and the Bendix Advanced Technology Center.

In 1983, Dr. Griffin joined APL's Design Engineering Group as assistant group supervisor for engineering. He participated in the Technology Modernization Program and was instrumental in bringing to APL computer-aided tools for mechanical engineering. In 1985, he returned to Virginia Polytechnic Institute and State University as an associate professor in the Engineering Science and Mechanics Department. His current research in the area of composites involves mechanics, three-dimensional analysis, and finding new applications.



WILLIAM E. SKULLNEY (right) was born in Blossburg, Pa., in 1953. He received both B.S. and M.S. degrees in engineering mechanics from the Pennsylvania State University. He joined APL's Strategic Systems Department as an engineer in the Launcher/Ship Group in 1978 and joined the Structural-Mechanical Design Group in the Space Department in 1981. Since then, Mr. Skullney has been involved in structural analysis and environmental testing of spacecraft and spacecraft components. He has performed such tasks on the Hopkins Ultraviolet Telescope and the HILAT spacecraft, and is involved with the Delta 180 program. Mr. Skullney used the finite-element analysis program MSC/NASTRAN in his analysis work.

JOHN A. ECKER's (left) biography can be found on p. 259.