APPLICATIONS OF GRAPHITE COMPOSITES TO DIMENSIONALLY STABLE SATELLITE STRUCTURES

JACK PRUNTY & DENNIS DUNBAR

GENERAL DYNAMICS CONVAIR DIVISION

SAN DIEGO, CALIFORNIA

ABSTRACT

This paper presents an appreciation of the success attained in the application of graphite composites to dimensionally stable satellite structures. To exemplify this, the discussion describes the performance attained in an early development article, examples of the subsequent proliferation of applications, and the accepted potential of graphite composites for the ultra-large satellite structures of the future.

INTRODUCTION

The graphite composite materials considered in this paper consist of graphite fibers embedded in thermosetting or thermoplastic matrices. Epoxy is currently the most commonly used thermosetting matrix material and polysulfone the most usual thermoplastic. For the applications treated here, the composite is made in laminated form by the successive layup of pre-impregnated unidirectional tape or pre-impregnated woven cloth. In either case, the directionality of the fibers is usually varied in steps through the thickness of the laminate to obtain the required properties.

These composites have gained wide acceptance in the aerospace industry as outstandingly efficient structural materials for a large variety of applications. Some noteworthy examples of such applications are discussed later.

In the realm of structural materials, graphite composites offer the unique capability of providing nominally zero thermal expansion or, alternatively, thermal expansion characteristics tailored to specific values. It is, of course, this attribute which has led to the adoption of these materials for a variety of dimensionally stable satellite structures. Before describing a selection of these applications, a brief discussion of some of the practical design considerations is pertinent.

In the case of theoretically zero-expansion laminates, and also in the case of those designed to exhibit a specified coefficient of thermal expansion (CTE), some deviations from the design value occur in the actual hardware. For potential applications, the attainable values are of significant interest. Tables 1 and 2 present data obtained in the Convair precision dilatometer for test coupons extracted from large structural panels. These values were attained in low-cost structure programs using routine techniques for manufacturing and material control, without attempting to control parameters specifically for the attainment of greater precision in the CTE.

These data, typical for "as manufactured" structural components, demonstrate that a CTE of 0 ± 0.10 microinch/inch °F is easily attained. This value is adequate for most applications, but could be improved to meet more stringent criteria by control of the material and processing parameters which affect the CTE.

Another significant advantage of graphite composites is given by the superior stiffness/density ratio obtainable in laminates using fibers of the high modulus or ultra-high modulus type. With a typical density of 0.060 lb/in^3, laminates with properties essentially isotropic in the plane of the laminate can exhibit a value of modulus/density of 250 x 10^6 inches compared to a range of 100 to 112 x 10^6 for the structural metals: aluminum, steel, and titanium. Obviously, for stiffness-critical structures, a weight reduction approaching 60% can be obtained as compared with an equivalent metallic structure. Such significant savings have been achieved in some of the applications discussed later. In many

Table 1. HEAO-B CTE test results.

Material: GY70/X30	
Layup: $(0/45/90/135)_{2S}$	
Specimen	"Isotropic" CTE (μin./in.°F)
	T = 68F
1	-0.050
2	-0.034
3	-0.019
4	+0.096
5	+0.064
6	-0.076
7	+0.028
8	+0.082
Avg	+0.011

Table 2. GEMS CTE test results.

Material: Modmor I/X30	
Layup: $(0_3/\pm45/90)_S$	
Specimen	Longitudinal CTE (μin./in.°F)
	T = 35F
1	-0.057
2	-0.049
3	-0.003
4	-0.004
5	-0.022
6	-0.017
7	+0.003
8	-0.056
9	-0.031
Avg	-0.026

strength-critical applications, dramatic weight reductions can also be obtained with graphite composites using suitable high-strength fibers.

A characteristic of the graphite composites which must be considered for dimensionally stable applications is hygroscopy: the material readily absorbs moisture from the atmosphere. This absorption induces a dimensional change which can be as high as 20 microinches/inch in the plane of the laminate for typical manufacturing conditions and time spans. Desorption can occur in the operational vacuum environment, with an attendant reversal to the original dimensions. Some dimensionally stable instruments, such as the Space Telescope (ST) have adjustment capabilities which can accommodate this change. For other applications where adjustment is not feasible, proven techniques are available for preventing moisture absorption by sealing of the surfaces.

Finally, in concluding this brief review of the dominant design considerations, it may be noted that the nonductile nature of the material is not reflected in unduly brittle characteristics, resistance to damage is adequate for most applications, and fatigue resistance is generally excellent.

The following sections will describe some development work in support of current applications, some actual current applications, and finally some dramatic applciations planned for the near future.

APPLICATIONS

The Space Telescope

The Space Telescope (Figure 1) is a 2.4 meter (94.5 inch) diameter reflecting telescope scheduled for launch into a 500 km (311 mile) earth orbit in 1983. The project is sponsored by the National Aeronautics and Space Agency (NASA). Recently, the contract for the design of the telescope was awarded to the Perkin-Elmer Corporation, Danbury, Connecticut. The discussion here concerns development work previously accomplished by General Dynamics Convair Division to demonstrate that a graphite-epoxy structure for the telescope could satisfy the stringent dimensional stability criteria and provide structural adequacy in all respects. This technology development program was performed under contract NAS8-28201 to the George C. Marshall Space Flight Center (MSFC).

The configuration of the telescope at the time of the technology contract award is shown in Figure 1. This configuration differs from the current version, primarily in diameter. The subsequent changes, however, do not invalidate the results of previous structural development. In the MSFC contract, General Dynamics Convair designed, fabricated, and tested a half-scale model of the metering structure using Modmor I/X30 graphite-epoxy.

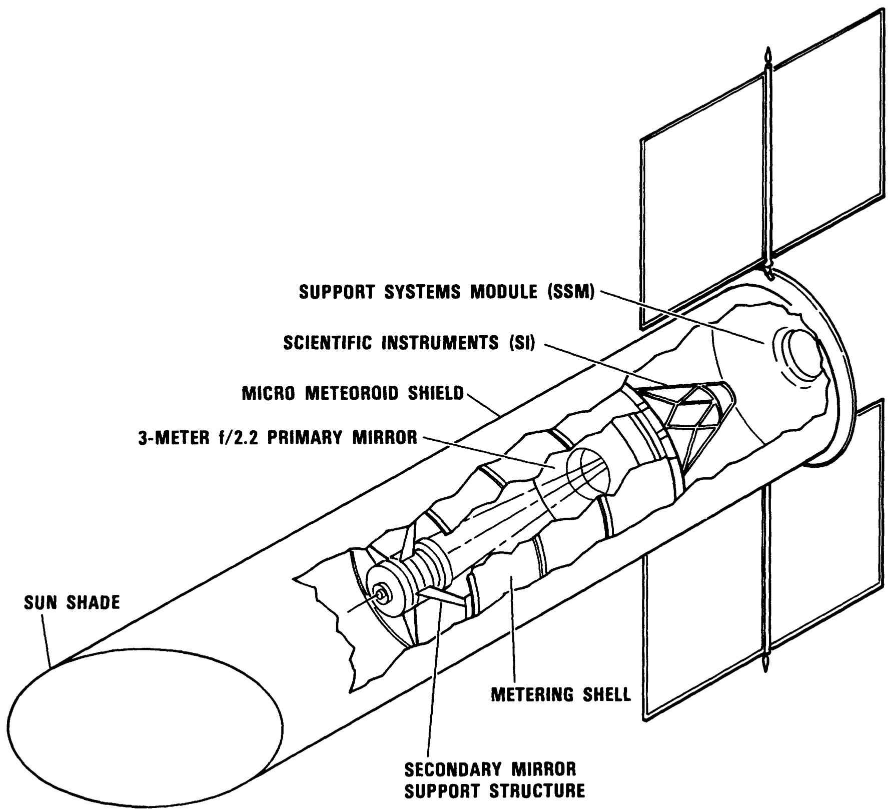


Figure 1. The Space Telescope (Phase A version).

The metering structure is visible in Figure 1 as the cylindrical "body" of the telescope between the primary and secondary mirrors. As it is seen in this view, the metering structure is enclosed in a cylindrical aluminum micrometeoroid shield. The primary function of the metering structure is to precisely maintain the spatial relationship between the primary and secondary mirrors. The secondary mirror is provided with focus, tilt, and centering adjustment but, once this adjustment is performed, it must be maintained within specified close tolerances throughout any period of observation. At the start of the technology program, the specified tolerance budget for the full-scale metering structure was:

Despace = ± 2.0 micrometers (±78.8 microinches)

Decenter = ±10.0 micrometers (±393.7 microinches)

Tilt = 4.9 microradians (1.0 arc second)

The 5.9m (232 inch) long metering structure was required to maintain the secondary mirror relative to the primary mirror within these permissible deviations during 5.55°K (10°F) structural temperature change. A little arithmetic shows that a material having a CTE not exceeding 0.034 microinch/inch°F is required to attain this dimensional stability. Structural metallics with values for the CTE ranging from 5 to 12 microinches/inch°F are completely ruled out. In contrast, as a structural material, graphite-epoxy has a unique capability of meeting these requirements.

Having established the suitability of graphite composite for this application, a study was made of the alternative shell and truss concepts for the metering structure. The conclusion of this was that both could meet the stability criteria; the truss would be lighter, but the cost of the shell structure would be lower. Since the latter criterion was the driver, with the concurrence of MSFC a decision was made to develop a shell-type structure. Figure 2 is a photograph of the half-scale metering structure which was subsequently designed and built. The GEMS (Graphite-Epoxy Metering Shell) is 2.95m (116.1 inches) long and 1.65m (64.68 inches) in diameter. As seen in the photograph, it is a simple semi-monocoque structure with a central secondary mirror support hub attached to the shell via four legs. The composite laminates were designed to exhibit theoretically zero CTE in the critical longitudinal direction. During fabrication, specimens were extracted from the shell skin panels to verify by test the attainment of an acceptable value for the CTE. As indicated in Table 2, the results were well within the allowable value of 0.034 microinch/inch°F.

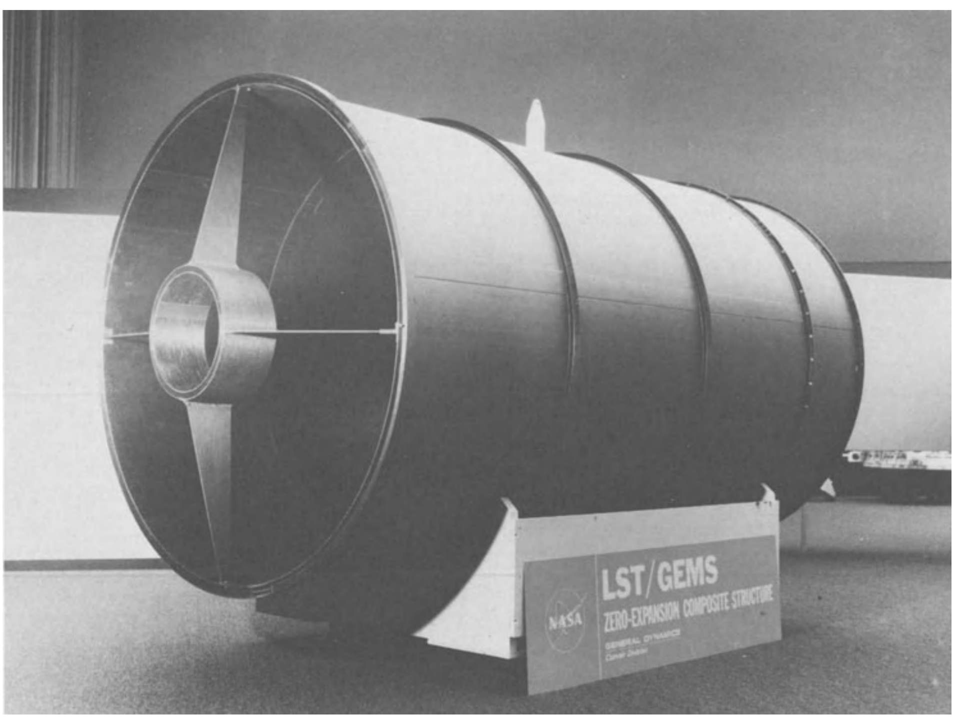


Figure 2. The GEMS Metering Structure.

The completed GEMS structure was subjected to thermal-vacuum chamber tests to demonstrate the overall thermal expansion characteristics. An important aspect of this program was the demonstration of a technique for measuring microinch strains in a large structure in a practical test environment. An adaptation of the basic Hewlett-Packard laser dilatometer measurement system was developed for use under these test conditions. Figure 3 shows a schematic of this system, as used for the simultaneous measurement of the defocus and tilt modes. The laser unit was located outside the vacuum chamber to shoot through an optical quality glass window. Inside the chamber two mirrors registered on the GEMS structure, one at each of the telescope mirror reference planes. Essentially, the laser apparatus used the interferometric fringe-counting technique to measure changes in the distance between the laser optical assembly and each of the reference plane mirrors. The unit then extracted the difference between these two measurements to determine the change in the spacing between the two reference mirrors. The changes in length of the GEMS structure were then displayed in digital form and recorded on tape to a resolution of 0.5 microinch.

During the test, the GEMS was cycled between -20°F and -100°F. Since the defocus requirement was calculated to be the most stringent, the following discussion will consider this mode. Fourteen thermal cycles were made between the structural temperature extremes, and defocus measurements were obtained for each cycle. The expansion characteristics were somewhat erratic over the first few cycles, a phenomenon which also occurs with graphite-epoxy in coupon-type testing. Dilatometer coupons are routinely subjected to ten cycles to stabilize the

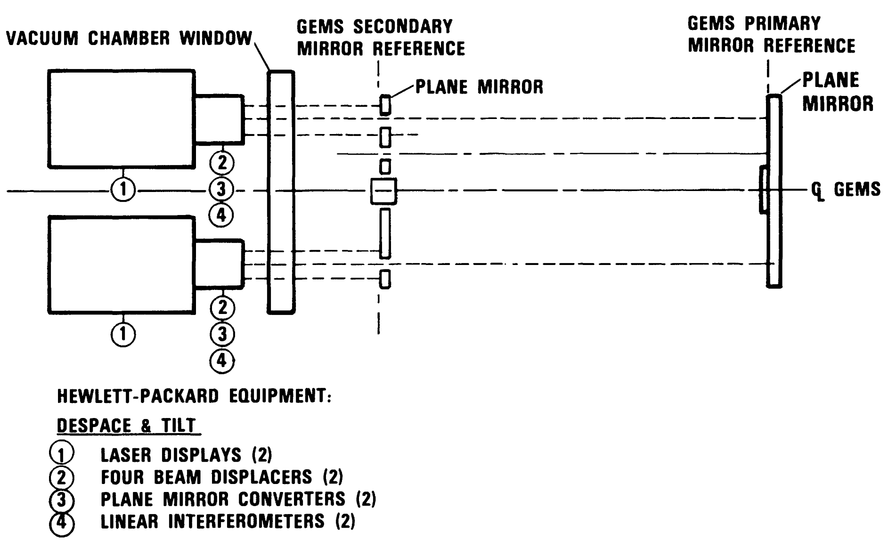


Figure 3. The Laser Dilatometer.

expansion characteristics prior to taking measurements, and complete structures are now similarly cycled prior to delivery. In the case of GEMS, a maximum value of thermal expansion of 1.88 micrometers/10°F occurred on the first cycle. A value of 1.52 micrometers/10°F was obtained on the fourteenth cycle when the characteristics had essentially stabilized.

The measured overall expansion of 1.52 micrometers/10°F exceeded the 1.00 micrometers/10°F half-scale value of the original defocus criterion. Diagnostic test runs and analyses traced the discrepancy to the canted legs of the secondary mirror support. This support geometry introduces a link-mechanism-type magnifying effect if zero CTE is not attained for the legs or for the associated support ring. An outcome of these findings was the adoption for the Space Telescope of a symmetrical support structure to correct this problem.

Considerable structural testing was also performed on the GEMS. During the test history, it was subjected to limit load some 50 times without encountering problems. Dynamic tests demonstrated a first mode fixed-base resonant frequency equivalent to 19 Hz for the full-scale metering structure. This value, which could be raised to 25 Hz by the stiffening effect of additional bolts in the attachment flange, exceeded the 15 Hz specified requirement.

Additional tests were performed with the cooperation of MSFC, in parallel with the GEMS program, to demonstrate compatibility of the structure to the orbital environment. First, the basic X30 resin system was qualified by MSFC tests to the outgassing requirements of NASA Document 40M51264. In addition, specimens were prepared which simulated the graphite-epoxy metering shell, the surrounding multi-layer insulation, and the outer aluminum micrometeoroid shell. Micrometeoroid impact tests by the Engineering Physics Branch at MSFC resulted in only roughening of the surface of the graphite-epoxy by the critical particle mass for the ST, and a two-inch diameter hole with only peripheral delamination by three times the critical particle mass.

The rather comprehensive testing in support of the ST did much to promote the use of graphite-epoxy for satellite structures, and lead directly to other applications described in the following sections.

HEAO-B Structures

The HEAO-B is the High Energy Astronomy Observatory, B Mission, an x-ray experiment scheduled for earth-orbital operation in 1978. Figure 4 depicts the HEAO-B satellite, which carries a powerful x-ray telescope coupled to a comprehensive array of instruments. The experiment is designed for detailed analysis of a great number and variety of galactic and extra-galactic x-ray objects. The forward end of the telescope can be seen in the figure as the large ellipse, surrounded by the three aspect sensors and the monitor proportional counter.

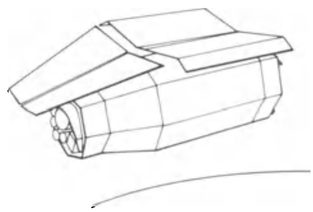


Figure 4. The HEAO-B satellite.

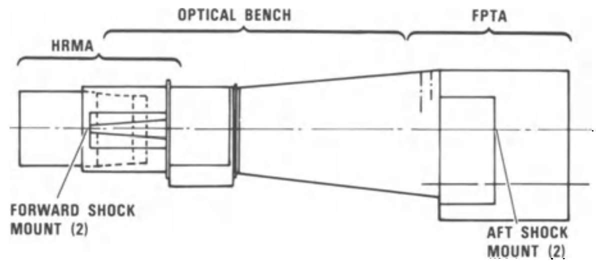


Figure 5. The HEAO-B telescope.

Figure 5 shows the general arrangement of the telescope. Attached to the forward end of the optical bench is the High Resolution Mirror Assembly (HRMA), a grazing incidence collector composed of concentric fuzed quartz cylinders. At the aft end, the optical bench is attached to the Focal Plane Transporter Assembly (FPTA) which houses a rotating array of instruments.

Graphite-epoxy was selected as the most suitable material for the optical bench and for the backbone structure of the HRMA. These components, which are described in the following sections, were designed and fabricated by General Dynamics Convair Division under subcontract to American Science and Engineering of Cambridge, Massachusetts.

HEAO-B OPTICAL BENCH — The completed optical bench is shown in Figure 6. It is a ring-supported, semi-monocoque structure, 119.12 inches long and 41.2 inches maximum diameter. The shell of the bench is constructed from four laminated panels joined by splice straps bonded with EA934 room temperature curing adhesive.

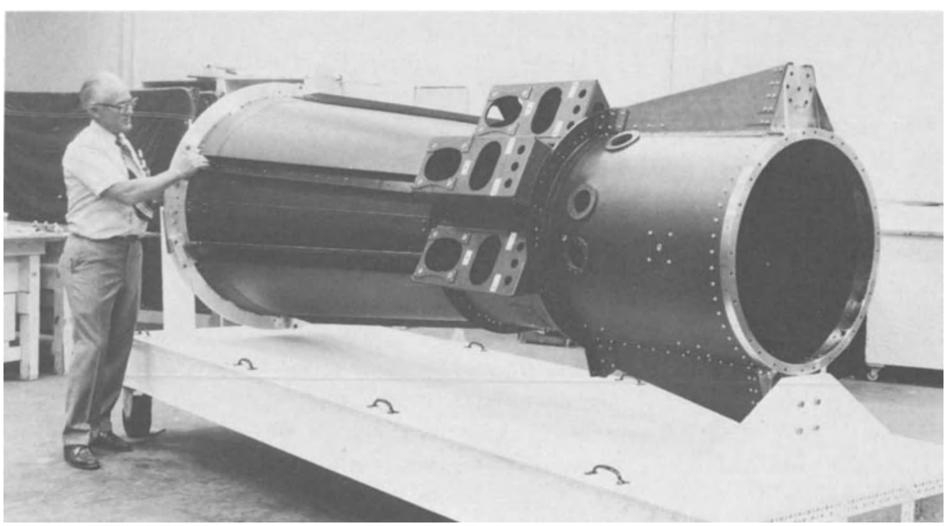


Figure 6. The HEAO-B optical bench.

The dominant design criteria, the first of which dictated the use of graphite-epoxy were:

- CTE longitudinally and circumferentially = 0 ± 0.10 microinch/inch°F
- Shell thickness — modulus product in both directions = 1.5×10^6 lb/in

These criteria were satisfied by a 0.104-inch thick, 16 ply, GY70/X30 laminate of $(0/45/90/135)_{2S}$ configuration. Ultra-high modulus GY70 fiber was used to meet the stiffness criterion for minimum weight and to attain a CTE isotropic in the plane of the laminate, within the specified range. The X30 resin was selected on the basis of freedom from microcracking, a phenomenon which can drastically affect thermal expansion characteristics.

It is noteworthy that, even with a laminate designed primarily for nominally zero thermal expansion, an elastic modulus 50% higher than the value for aluminum was attained with a material 36% less dense. This indicates that the weight of the basic shell structure is 43% of the weight of an aluminum structure designed to the same stiffness criterion. Compliance with the stiffness criterion was demonstrated by tests of three coupons from each of the four skin panels. The test values for the tensile modulus ranged from 15 to 17×10^6 psi. As shown in Table 1, compliance with the specified CTE was also demonstrated by dilatometer tests of a longitudinal and transverse coupon extracted from each of the four skin panels.

A feature of interest in this application was the adherence to stringent dimensional tolerances on the optical bench assembly. Exemplifying this are the three aspect sensor mounts, conspicuous in Figure 6, which were maintained within ± 0.005 inch relative to the forward face of the bench and the optical axis.

An additional point of interest is that the NASA permitted a protoflight approach for the optical bench. In this approach to the design of flight hardware, a factor of safety = 3 is used, structural integrity is demonstrated by analysis only, and all requirements for structural testing are eliminated. This curcumstance demonstrates the high degree of confidence which has been established for this type of graphite-composite structure.

HRMA CYLINDERS — These structural components, which support the concentric fuzed quartz cylinders in the mirror assembly, are shown in Figure 7. The cylinders vary in size from 10.740 inches diameter by 20.955 inches long to 25.410 inches diameter by 20.615 inches long.

Four prime design requirements for the cylinders were decreed to minimize distortion in the mirror assembly:

1. Attain a CTE of 0.3 microinch/inch°F longitudinally and circumferentially for compatibility with the thermal expansion of the fuzed quartz mirrors.

Figure 7. The HRMA cylinders.

2. Comply with specified shear and bending stiffness requirements.
3. Eliminate dimensional changes due to moisture.
4. Maintain geometrical tolerances on the outer surface of ± 0.005 inch on diameter and within 0.003 inch concentricity.

The CTE requirement could have been met with LR-35, a version of INVAR low-expansion nickel alloy. A comparison of the modulus/density ratios, however, indicates that INVAR cylinders would weigh 2.4 times as much as graphite-epoxy versions for equal stiffness. As in the case of the optical bench, ultra-high modulus GY-70/X30 graphite epoxy was selected to enable compliance with the stiffness requirements for minimum weight. The laminates evolved by the analysis were 0.572 inch thick, 88 ply, $(0_8/90_{12}/\pm 50_{12})_S$ for the outer cylinders, and 0.286 inch thick, 44 ply, $(0_4/90_6/\pm 50_6)_S$ for the inner cylinders. (This simplified notation does not indicate the actual stacking sequence of the plies.)

As shown in Figure 7, the cylinders were encapsulated in aluminum foil to inhibit moisture effects. Actually, a simple deposition coating is available for this purpose, but at the time of the cylinder fabrication the facility was not of adequate size. The technique adopted had been previously demonstrated as a means of eliminating moisture by long term dilatometer testing for moisture-induced dimensional changes. This sealing technique consists of applying, by adhesive bonding, a double layer of 0.0007-inch-thick aluminum foil with a schedule of initial, intermediate, and final dryout cycles.

Close control of the outer surfaces was attained by curing the cylinders in female molds with a tolerance of ± 0.003 inch on the mold diameters. The precision molding technique developed for this purpose is now finding a variety of applications. Also, the general development of this type of graphite/composite cylinder creates the possibility of providing lightweight substrates for the mirror cylinders of future very large x-ray telescopes.

The preceding examples give an appreciation of the successful exploitation of graphite-epoxy to meet demanding requirements in current applications. The following sections describe some of the more exciting and challenging applications planned for the future.

Large Space Structures

With the advent of Space Shuttle, space operations are entering a period of transition from single missions and limited-duration activities to continuous operations embracing a wide spectrum of applications. A new dimension in space structures is being introduced with extremely large, ultra-low-density arrangements required for such applications as satellite power systems, phased-array antennas, microwave radiometry arrays, and large platforms providing common structural and operational support for a variety of experiments and operations. A topically pertinent example is the solar power satellite (SPS) concept shown in Figure 8, an immense satellite structure which can significantly augment electrical energy supplies at a time when many terrestrial energy sources are approaching depletion. Such enormous structures require assembly in space because of payload volume and weight limitations of the Shuttle and even of future heavy lift launch vehicles. Graphite composites are expected to play a dominant role in this challenging field. General Dynamics Convair is engaged in development in this area under contract to the NASA Johnson Space Center and the Marshall Space Flight Center.

Critical considerations for such large orbital arrays are thermal and dynamic response. Cumulative distortions due to thermal gradients in large arrays could cause major transient geometric changes that would degrade the performance of space systems, and initially create acute problems in the in-space assembly of ultra-large structures. The magnitude of the potential thermal deformation problem is demonstrated in Figure 9. This figure shows, for various materials and surface characteristics, the thermal deformation of a 200m long space structure beam with two beam caps exposed to solar heating and one cap shadowed. The advantage of graphite composite materials for thermal distortion control is evident.

NASA and industry are studying the problems involved in transferring such large satellite structures from the ground to orbit. Two concepts showing promise are described here. Each has a role to play, depending on the ultimate size of the structure and the time frame when the structure is required.

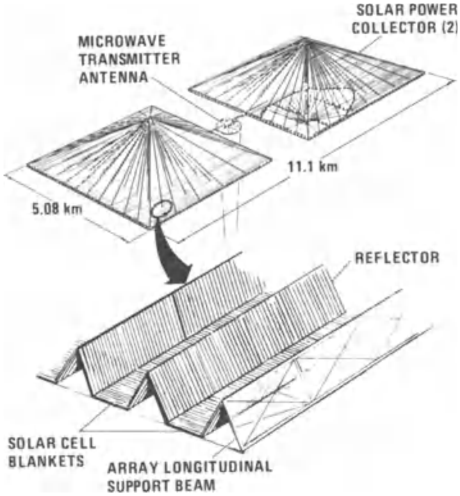
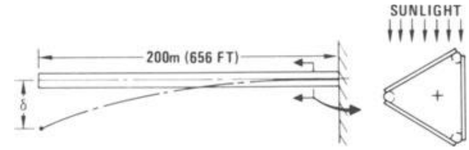


Figure 8. The Solar Power Satellite.



MATERIAL	SURFACE	CTE (PPM/°F)	ΔT MAX (°F)	TIP DEFLECTION δ (m)	(IN.)
ALUMINUM	WHITE	12.00	103	16.45	647.6
GRAPHITE (ISOTROPIC)					
AS/P-1700	WHITE	1.42	103	1.95	76.8
AS/P-1700	BARE	1.42	252	4.78	188.0
GY70/P-1700	WHITE	0.10	103	0.14	5.0
GY70/P-1700	BARE	0.10	252	0.32	11.8

Figure 9. Graphite composites eliminate critical distortion problems.

1. Deployable Structures — Folded and packaged modular elements can be carried in the shuttle, then deployed into approximately 100 meter building blocks. Such building blocks can be assembled in orbit into larger structures.
2. In-space Fabrication of Structures — Automated fabrication of very large structures from raw tape material can be accomplished in orbit using a "beam builder" machine. This concept is geared for the longer term, kilometer size, structures such as Solar Power Satellites.

DEPLOYABLE STRUCTURES — The deployable structure concept is based on a fundamental geometric shape, the tetrahedral truss, which has been so designed as to be foldable into a closely packaged element. The deployable tetrahedral truss was the outcome of studies performed at General Dynamics Convair during the late 1960s. Since that time, concept development has continued through several major NASA study contracts. In 1970, General Dynamics was awarded the patent on the concept with a provision for full usage rights by NASA.

When designed for large space strucures to be carried up in the Shuttle, the deployable tetrahedral truss is packaged as a 4.4 meter diameter module 6 meters long. Three such packages can be carried in the Shuttle as shown in Figure 10. Each package deploys into a hexagonal structure 335 feet by 290 feet. These structures can be joined in orbit into a single large satellite. Even larger structures can be assembled, using this modular approach with multiple shuttle launches. Launch costs, however, suggest that transporting material in a denser form — allowing more structure per launch, is desirable for extremely large structures such as Solar Power Satellites. This consideration resulted in the concept of in-space fabrication.

IN-SPACE FABRICATION — This concept of manufacturing structural beams in orbit will be exploited for many and various applications. These will range from moderately large sized structures to the tens of square kilometers size category required for the Solar Power Satellite.

At the heart of this concept is the beam building machine, which will automatically fabricate a basic structural beam from reels of preconsolidated graphite-thermoplastic strips. This basic beam, shown in Figure 11, is of triangular cross-section and open-truss configuration. A semi-schematic of the beam builder is presented in Figure 12. As shown in this figure, the graphite-thermoplastic strip material is supplied to the machine on reels. The feasibility of using the beam builder to fabricate very large structures is illuminated by the fact that one 14-foot diameter reel, as sized to the constraints of the Space Shuttle payload bay, will hold sufficient strip material for 17 miles of a structural element! A total of six reels is carried by the beam builder — one for each of the three cap elements and one for each of the three side-bracing arrangements. In the fabrication process, the strip material for each structural element is first heated to the forming temperature, then rolled to the required cross-section and subsequently cooled to regain rigidity prior to the assembly procedure. In the assembly stage, the side bracing members are cut off, automatically manipulated into position relative to the beam caps, and then attached to the caps by ultrasonic welding of the thermoplastic material. The basic forming process, a patent of General Dynamics Convair Division, is known as Rolltrusion.

Future ultra-large, very low density structures can utilize the basic truss beam for the caps and bracing members of beams an order of magnitude greater in size. Development of in-space fabrication techniques is being vigorously pur-

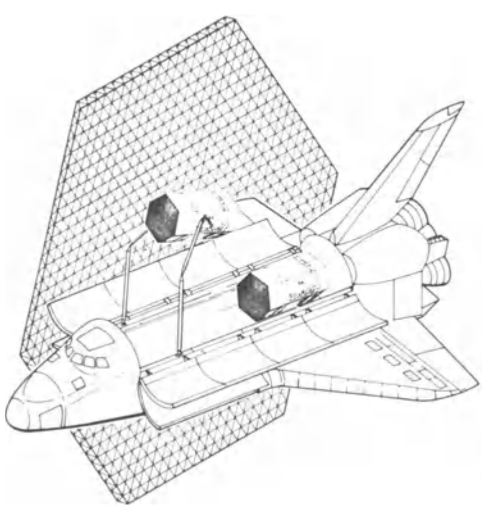


Figure 10. Deployable structure modules in the Shuttle.

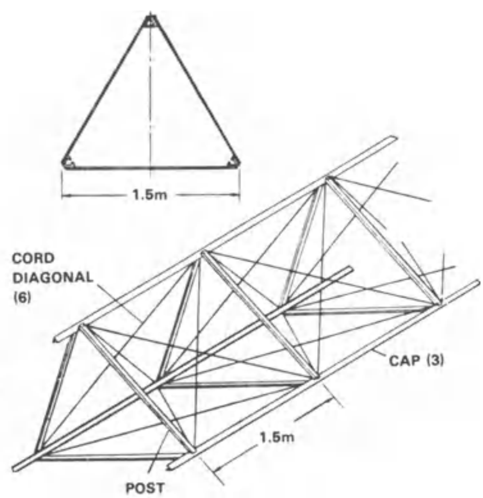


Figure 11. Beams employ lightweight, open-truss construction.

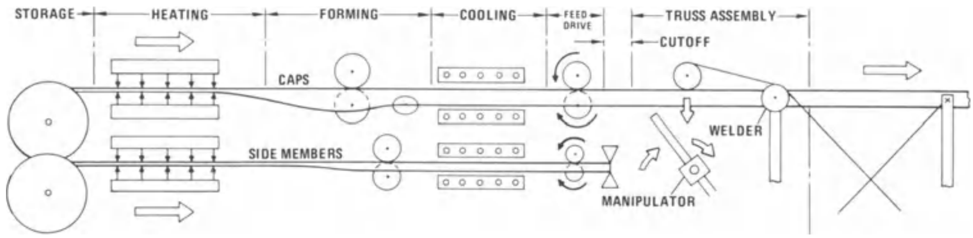


Figure 12. Semi-schematic of the beam builder (typical for three places).

sued by NASA and industry, and satellites of impressive dimensions should be feasible in the near future. To illustrate this, Figure 13 depicts a pioneering application of the beam builder. In this case, a single Space Shuttle mission, planned by the NASA JSC for 1982, will fabricate and assemble a 200-meter long satellite platform. This accomplishment will introduce an era of satellites of very large size, and will pave the way to the ultra-large orbital structures of the not too distant future.

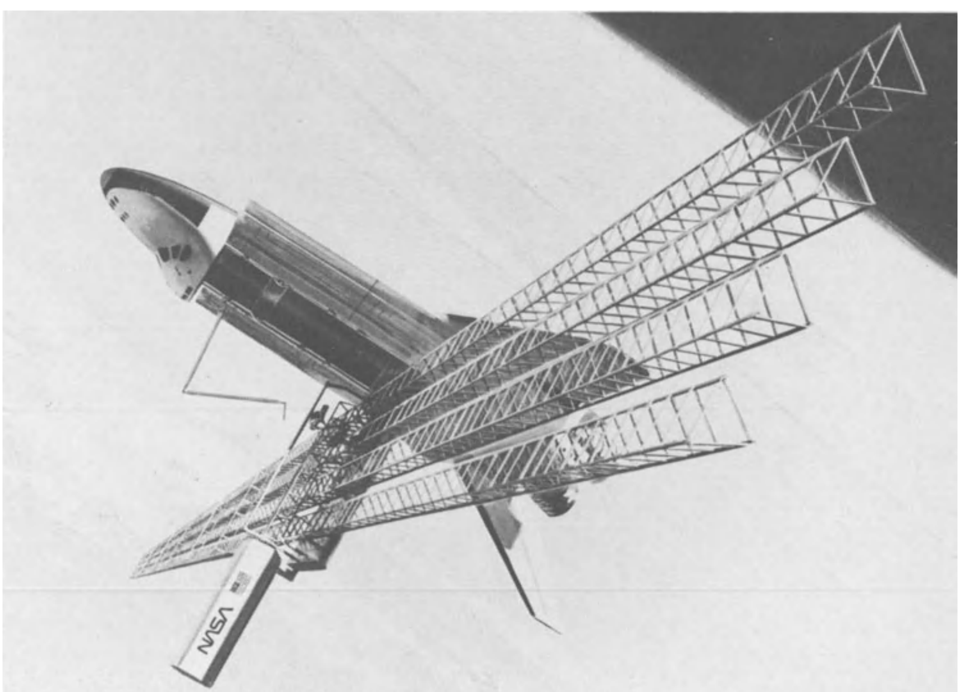


Figure 13. A single Space Shuttle mission fabricates and assembles a 200-meter long platform.

CONCLUSIONS

In conclusion, the application of graphite composite to large satellite structures is essentially current state-of-the-art. The adoption of composites for planned spacecraft structures can realize significant benefits in stiffness-critical and strength-critical applications and can be a virtual necessity where extreme dimensional stability is required. For the future, the unique characteristics of these high performance materials can offer great benefits to large space structure programs by contributing substantially to the minimization of total program costs and to the maximization of operational efficiency.

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The Smithsonian Astrophysical Observatory
The Massachusetts Institute of Technology
The Goddard Space Flight Center
The Columbia Astrophysics Laboratory

The experiment prime contractor is American Science and Engineering, Inc. The spacecraft contractor is TRW. The Space Construction Automated Fabrication Experiment definition study is under Contract NAS9-15310 to NASA JSC. Mr. Lyle Jenkins is the Contracting Officer's Representative.