N74 30939

THE BI-COMPOSITE TRANSITION JOINT

SD 72-SH-0051

K. C. DULLEA, JR.

MAY 26, 1972

Space Division
North American Rockwell

323<

THE BI-COMPOSITE TRANSITION JOINT

By K. C. Dullea, Jr.

The application of advanced composite materials to high performance structure frequently results in the desire to fabricate a structure from more than one composite system in order to tailor the composite material capabilities to the design requirements. The bi-composite transition provides a means of joining two different composite structural systems without the weight and complexity of mechanical attachments. The monolayer plies or combinations of plies of one composite system are interleaved with and bonded to the plies of the adjacent composite system, thereby providing a direct load transfer between the two composite structures. The bi-composite transition joint is suitable to general applications of composite materials and is not constrained to specific unique applications.

324<

BI-COMPOSITE JOINT DEFINITION

The term "advanced composite" is used to represent material combinations such as boron/aluminum, boron/epoxy or graphite/epoxy which are composed of high strength and high stiffness fibrous material integrated into a matrix to produce high performance structure. A significant feature of advanced composite laminates is that they can be tailored to the required design loads by lamina quantitive and cross plying build up.

The several available advanced composite material systems provide the composite designer with many different choices concerning design allowables and mechanical properties. There is a considerable variance in design allowables such as tension, compression and shear strength as a function of fiber volume percent as well as between material systems. Table 1 shows the variance between material systems for boron based composites of 50 percent fiber volumes and for graphite based composites with 55 percent fiber volume This data shows a compression strength allowable for boron/epoxy of 250 ksi, for graphite/epoxy of 140 ksi and for boron/aluminum of 157 ksi. Further investigation of the data reveals difference in tensile strength, tensile modulus, compression modulus and temperature effects for the noted composite material systems.

There are also choices from the mechanical properties such as density, thermal expansion coefficient, thermal conductivity, minimum form radius, machineability, ply thickness and cost as shown in Table 2. This mechanical properties data is based on the same fiber volume percent as the design allowable data. It can be seen that graphite based composites are less dense than boron based composites. The boron composites design allowables are generally higher than those for graphite composites, however because of the density difference the specific strengths are very close. The tensile strength factor for graphite epoxy is 3.04 and for boron/epoxy is 2.44. The minimum form radius for boron composite is 0.50 inches while for the graphite composite it is only 0.06 inches. Many other differences can be observed among the referenced mechanical properties.

Table 1. Recommended Composite Design Allowables

Composite System		B/Al		B/E		B/PI		G/E, HS		G/E, HM		G/PI, HS		G/PI, HM	
Fiber Volume % Density lb/in.^{3}		50 0.096		50 0.072		50 0.074		55 0.056		55 0.058		55 0.058		55 0.060	
Temperature, F		RT	350	RT	350	RT	600	RT	350	RT	350	RT	600	RT	600
Tensile Strength, ksi	0°	157	157	178	138	170	130	170	165	105	95	150	140	80	80
	90°	11.5	11.5	8.6	5.4	8.6	5.4	5.0	3.5	4.8	3.0	2.2	0.85	2.2	0.85
Modulus, msi	0°	32	29	30	29	30	29	21.0	19.6	29.0	29.0	19.0	19.0	27	27
	90°	19.6	14.4	2.7	1.1	2.7	1.1	1.1	1.0	1.1	0.8	1.3	0.8	1.1	0.8
Compression Strength, ksi	0°	157	157	250	134	225	125	140	70	100	80	125	54	90	48
	90°	11.5	11.5	36	15	36	15	35	25	35	25	14	11.5	14	11.5
Modulus, msi	0°	32	29	28	28	28	28	21.0	19.6	27	25	17	17	22	22
	90°	19.6	14.4	2.8	1.5	2.8	1.5	1.5	1.0	1.5	1.0	1.1	0.85	1.1	0.85
In-Plane Shear Strength, ksi	0°	10.	10.	8.5	3.0	5.0	2.5	7.0	2.0	5.0	2.5	3.0	1.5	3.0	1.5
Modulus, msi	0°	6.4	6.4	1.0	0.6	1.0	0.6	0.81	0.64	0.81	0.64	0.81	0.64	0.81	0.64
Major Poisson's Ratio μ in./in.		0.22	0.24	0.20	0.24	0.20	0.24	0.25	0.3	0.24	0.3	0.37	0.51	0.37	0.51

Table 2. Advanced Composite Mechanical Properties

		B/Al	B/E	B/PI	G/E		G/PI	
					HS	HM	HS	HM
Density	lb/cu in.	0.096	0.072	0.074	0.056	0.058	0.058	0.060
Specific Tensile Strength, R.T.	10^6 inch	1.64	2.44	2.30	3.04	1.82	2.58	1.33
Specific Modulus, R.T.	10^8 inch	3.33	4.16	4.04	3.76	5.00	3.27	4.50
Maximum Operating Temperature	F	650	350	600	350	350	600	600
Thermal Expansion Coefficient	in./in./F	3.0	2.5	2.5	0	0	0	0
Thermal Conductivity	$\frac{\text{Btu/in.}}{\text{hr/sq ft/F}}$	565	14	14	100	100	100	100
Machinability		Poor	Medium	Fair	Very Good	Very Good	Very Good	Very Good
Bend Radius	in.	Large	.50	.50	.06	.06	.06	.06
Minimum Ply Thickness	in.	.0053	.0053	.0053	.003	.003	.003	.003
Cost	$/lb	260* 454*	300* 524*	667* 775*	247	270	278	301

327<

Because of the large selection of design allowables and mechanical properties available, in some design applications, it may become desireable to change from one advanced composite material system to a different one, i.e., boron/epoxy to graphite/epoxy. This approach is a continuation of the design flexibility already being used in aerospace structure advanced composite material system design applications. The bi-composite joint provides a means of joining two different advanced composite structural systems to tailor the unique design allowable and mechanical properties of each system to the design requirements of interest.

The bi-composite joint therefore is an interface between two dissimilar materials. Advanced composite structures are normally fabricated by means of tape layup where the plies of material are stacked in a predetermined pattern to provide the desired structural properties. In using bi-composite joint concept, a portion of the structure is fabricated from one composite material to take advantage of its properties while the remainder of the structure is fabricated of a different composite material to take advantage of other material properties. The bi-composite joint is the joining of these two composite systems as shown in Figure 1. Plies of one composite system are interleaved with plies of the second composite system and the voids are filled with epoxy resin. The bi-composite joint provides a smooth transition and transfer of loads across the joint.

The structural requirement in designing the bi-composite joint is to bond each layer of the first composite system to the corresponding layers in the adjacent composite system. This is required because of the cross plying used with composites. For example, with the layer orientation used in the illustration [0/0-/±45/∓45/90/0], the boron/epoxy outermost layers are arranged with the fibers oriented in the longitudinal direction. Therefore, the graphite/epoxy material must also be arranged with the outermost layers in the longitudinal direction. This approach provides an ordered load transfer across the joint, i.e., tension to tension, shear to shear, etc.

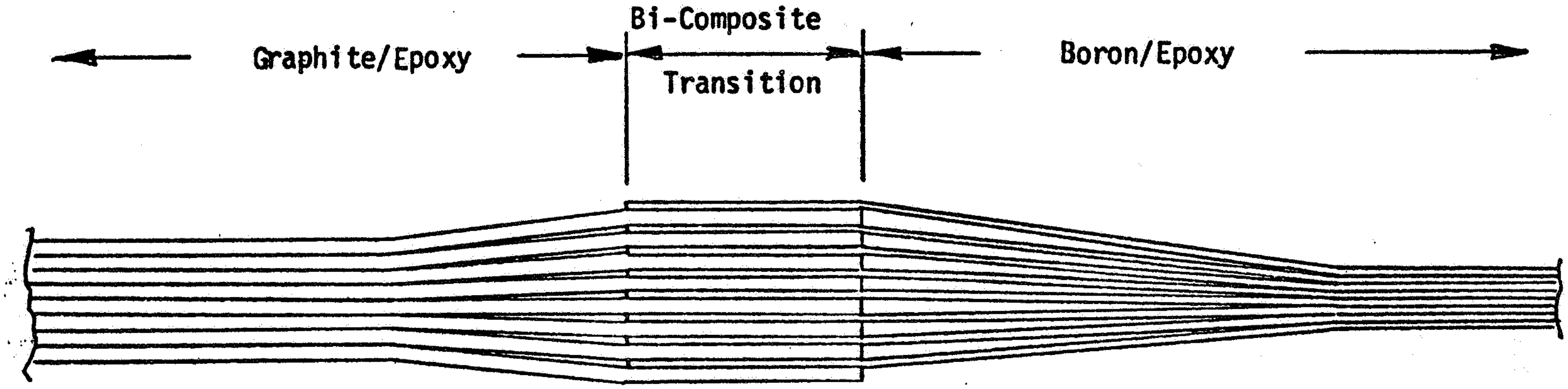

Figure 1. The Bi-Composite Transition Joint

REQUIREMENTS FOR BI-COMPOSITE JOINTS

One requirement for bi-composite joints involves the forming of a boron/epoxy stringer. The boron/epoxy composite system requires a 0.50 inch form radius which is difficult to layup and impractical to interface with the end fittings. One solution to the problem is to use a bi-composite transition joint and fabricate the end of the stringer from graphite/epoxy which has a 0.06 inch form radius as shown in Figure 2. This approach permits forming the end of the stringer into a closeout fitting that accepts mechanical attachments.

Another bi-composite joint requirement involves the efficiency of composite structures loaded in compression. Boron/epoxy has a twenty percent greater specific compressive strength than graphite/epoxy. However, as in the previous example, the boron/epoxy 0.50 inch form radius causes difficulty in fabricating hat section stiffeners for a skin stringer panel. The bi-composite joint can be applied by using graphite/epoxy layers arranged in [±45] orientation draped over the entire hat surface with boron/epoxy layers used on the caps only in the [0] orientation as shown in Figure 3. The resulting stringer has the advantage of the higher specific compressive strength of boron/epoxy and the smaller form radii available by using graphite/epoxy.

Thermal isolation requirements provide a use for the bi-composite joint. In the case of reentry vehicle with external thermal protection and advanced composite primary structure, the structure could be fabricated from graphite/epoxy or boron/aluminum to take advantage of lower material and fabrication costs with the thermal stand-offs fabricated from boron/polyimide to provide thermal isolation and higher temperature capability as shown in Figure 4. This requirement for the bi-composite joint permits the design of an all composite structure where the only mechanical joints are the attaching of the thermal protection panels.

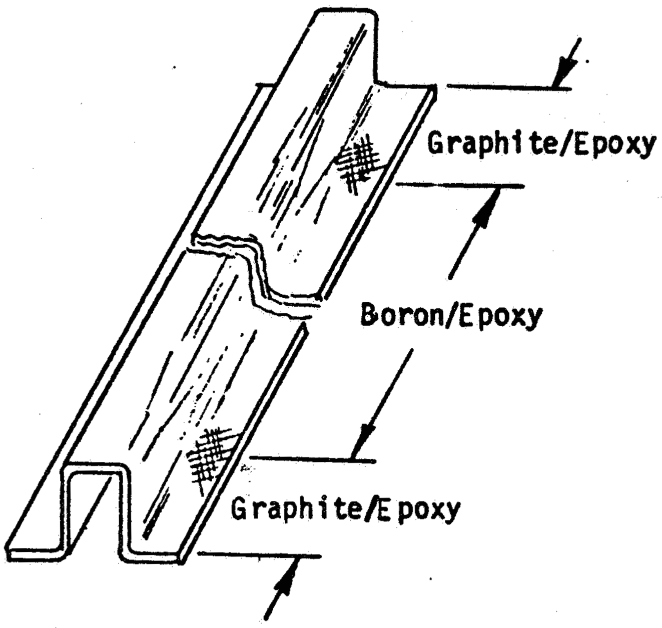

Figure 2. Bi-Composite Structural Reinforcement

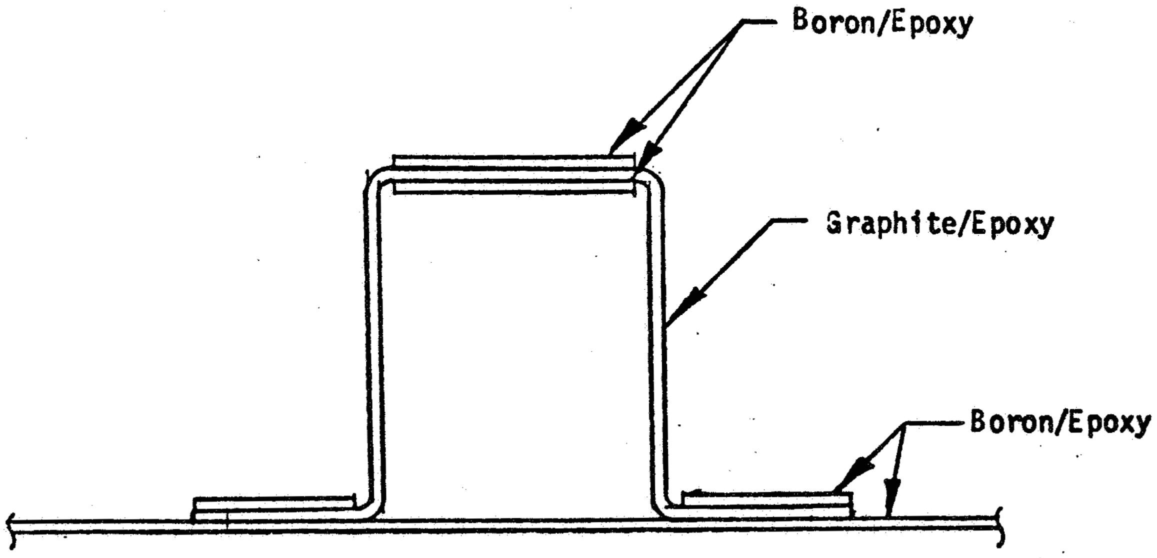

Figure 3. Bi-Composite Thermal Isolation

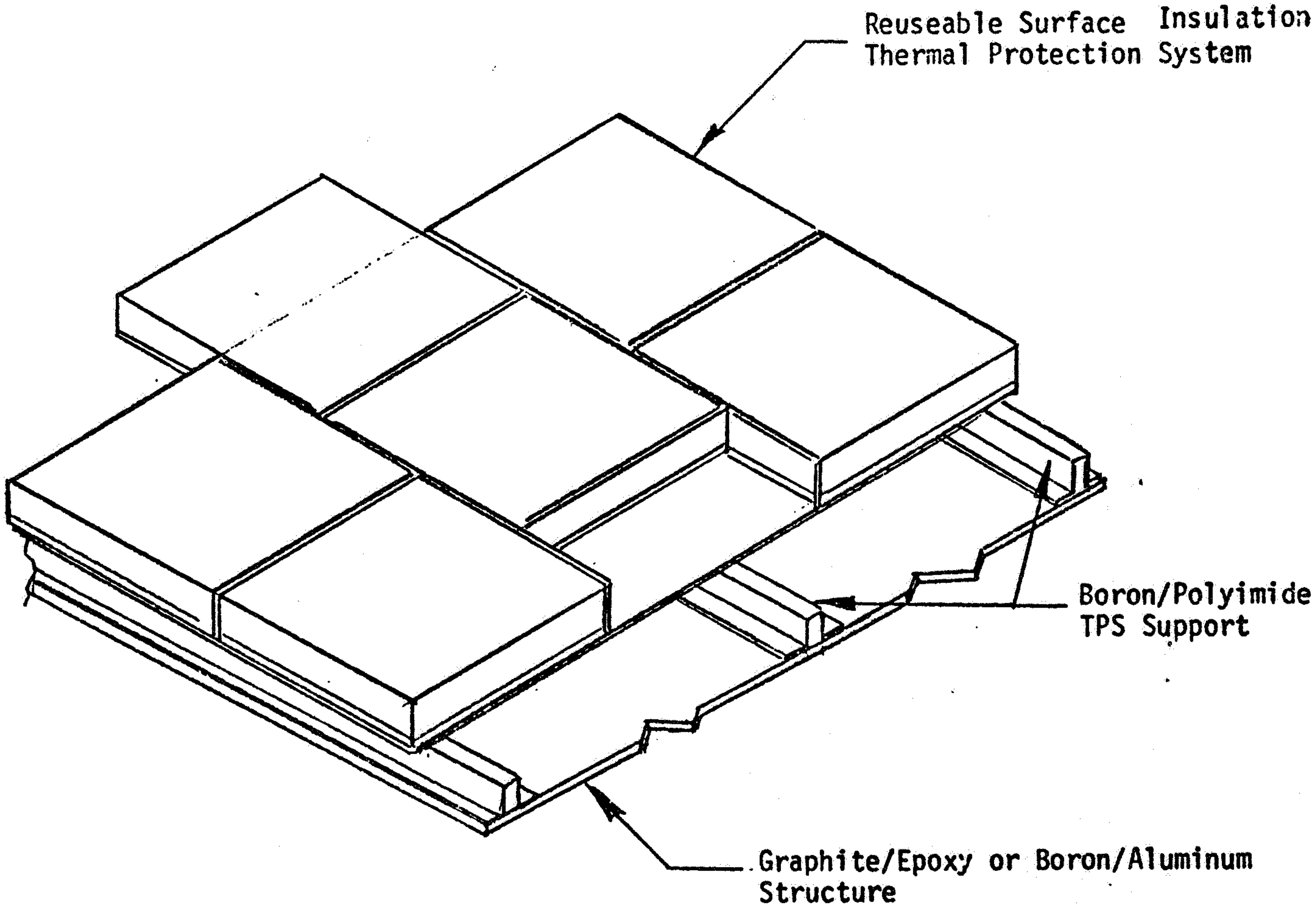

Figure 4. Bi-Composite Thermal Isolation

332<

Thermal distortion requirements also provide for application of the bi-composite joint. In designing advanced composite structures such as antenna disks, solar panel structures and thrust structures which are subjected to large thermal gradients, but where thermal deflection must be minimized, graphite based advanced composite systems should be used to take advantage of the very low, approximately zero, coefficient of thermal expansion. However graphite composites have a fairly high thermal conduction for non-metallics, therefore the supports for these items should be fabricated from boron composite materials to provide thermal isolation from the vehicle structure as shown in Figure 5.

One difficulty in applyinq boron fiber composite materials to aerospace structures is the drilling of holes for mechanical attachments. Successful use of mechanical attachments in composites has been demonstrated by interleaving metal shims for local reinforcement. However, drilling holes through a boron/epoxy laminate interleaved with metal shims is a very difficult operation. Again the bi-composite joint provides a means for designing around this difficulty by making a transition from boron/epoxy to graphite epoxy interleaved with titanium shims thereby providing a graphite/epoxy and titanium structure for drilling as shown in Figure 6. This joint permits a boron/epoxy composite structure with the resulting high compressive strength without the penalty in drilling and machining the boron material.

Combined stiffness and thermal warp requirements present another use for the bi-composite joint. On large manipulator arms such as those proposed for orbital payload handling, stiffness is a driving function. However, form radius and thermal warp make graphite composite necessary for the [±45] layer. The longitudinal layers of the structure could be made from a boron composite and significantly increase manipulator arm stiffness with a resulting small increase in thermal warp.

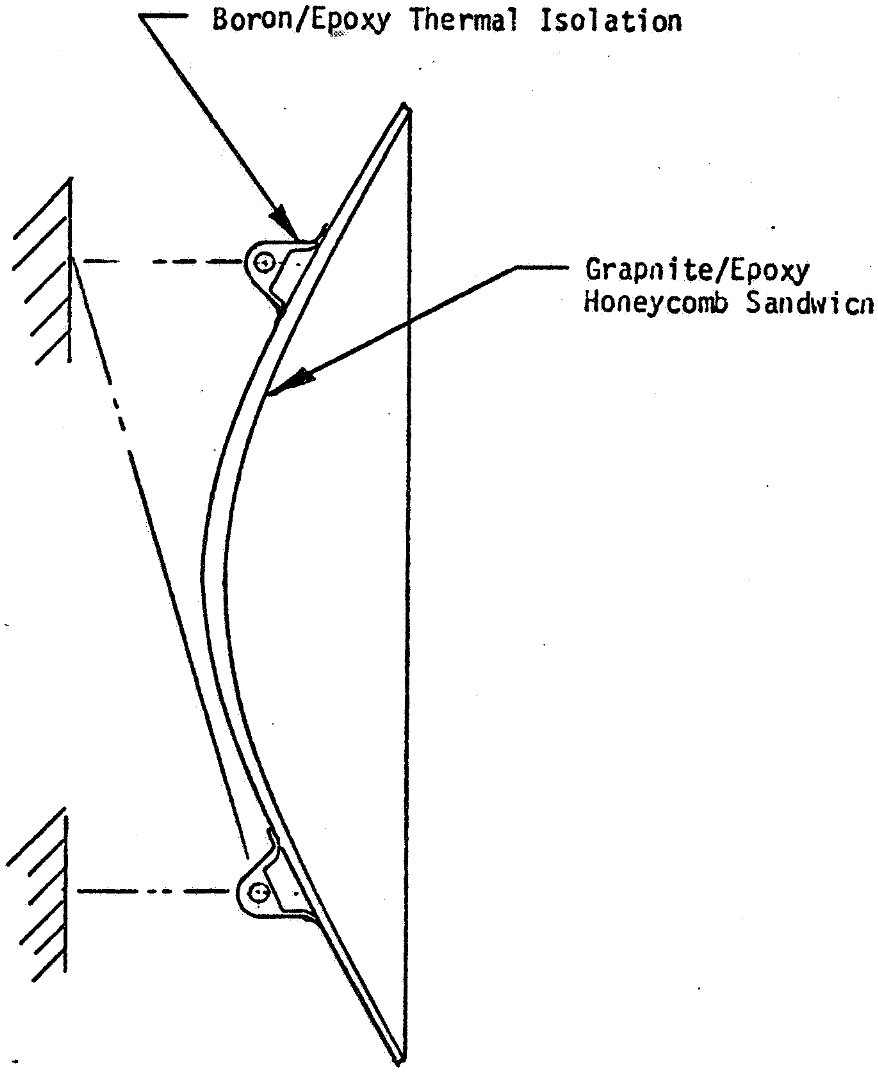

Figure 5. Bi-Composite Thermal Expansion

334<

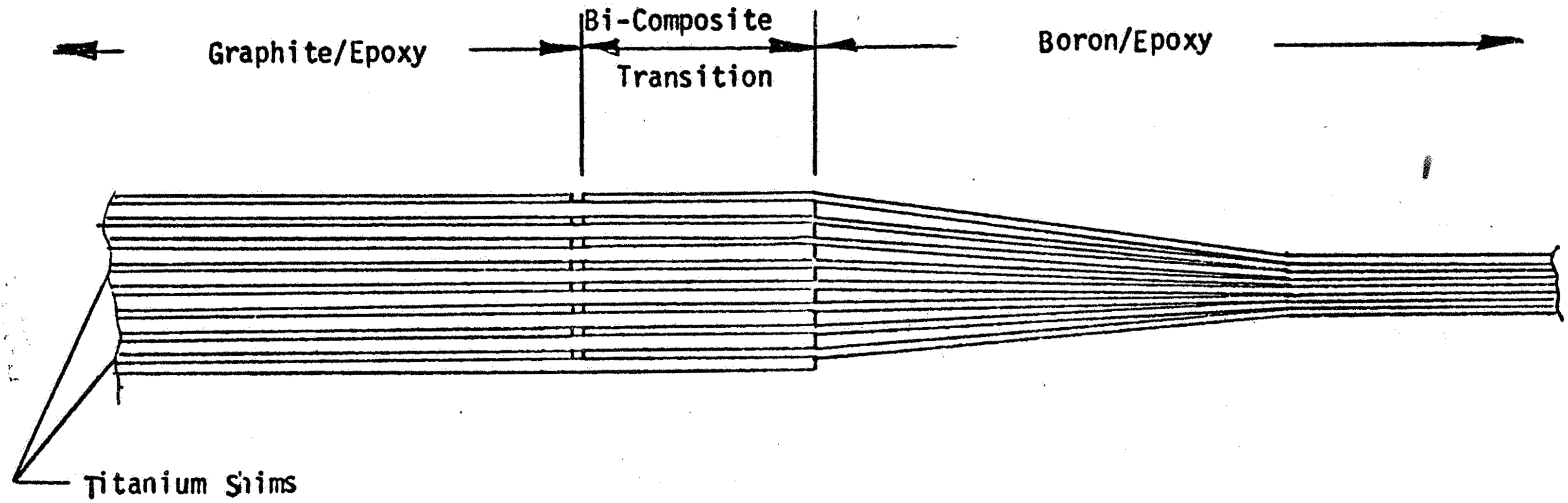

Figure 6. Bolted Joint Bi-Composite Requirements

TYPICAL BI-COMPOSITE JOINT APPLICATION

Several Saturn S-II stage unpressurized structures were considered for application of advanced composite materials in a recent NASA Contract NAS8-27278 with Marshall Space Flight Center. These structures, shown in Figure 7 include the forward skirt, aft skirt, thrust cone structure, interstage structure and the engine support longeron with the aft skirt being selected for advanced composite materials design application study. The composite applications included boron/epoxy and graphite/epoxy skin stringer structures as well as boron/epoxy and graphite/epoxy honeycomb sandwich as shown in Figure 8.

The aft skirt is a 33 foot diameter cylinder at the mold line, located by the inner surface of the skin, approximately 7 feet in length and stiffened by three ring frames. There are 216 extruded hat-section stiffeners, two inches high and two inches across the crown, equally spaced around the shell perimeter, resulting in a stringer center to center spacing of 5.76 inches.

The aft skirt is a load carrying shell structure, which transmits loads from the interstage during Saturn I/C boost and from the thrust cone during Saturn II boost to the aft flange of the LOX tank bolting ring. Two of the ring frames are dual purpose members providing shell stability for the aft skirt during Saturn I/C boost and thrust cone structural integrity during Saturn II boost. The maximum external body limit loads and associated distributed loads critical for aft skirt design are 5265 pounds per inch compression and 1250 pounds per inch tension. The aft skirt differential pressure is 3 psi and occurs at max $q\alpha$. The temperature extremes are 200 F on the interstage end and -280 F at the LO$_2$ tank interface.

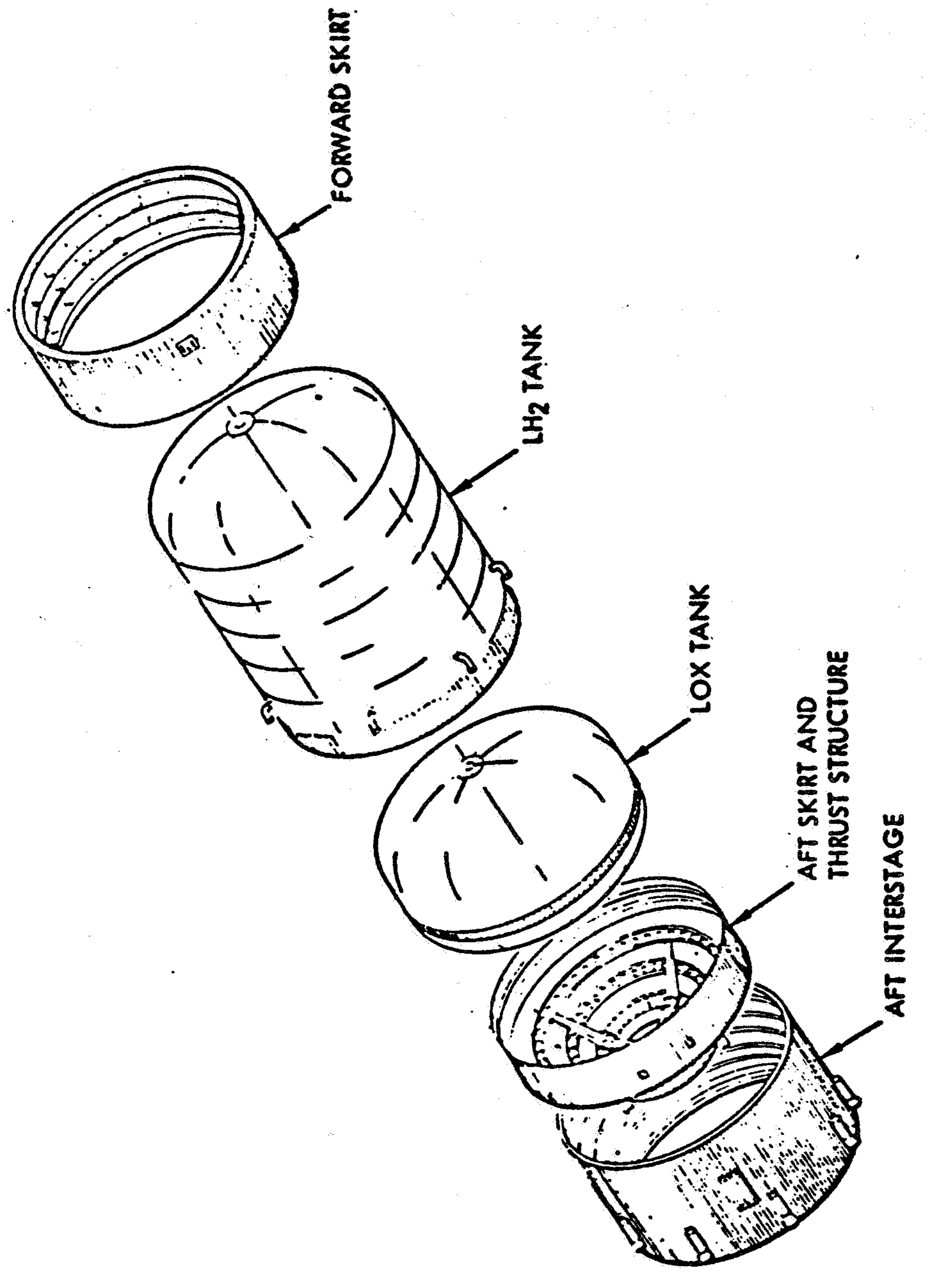

Figure 7. S-II Subassemblies

337<

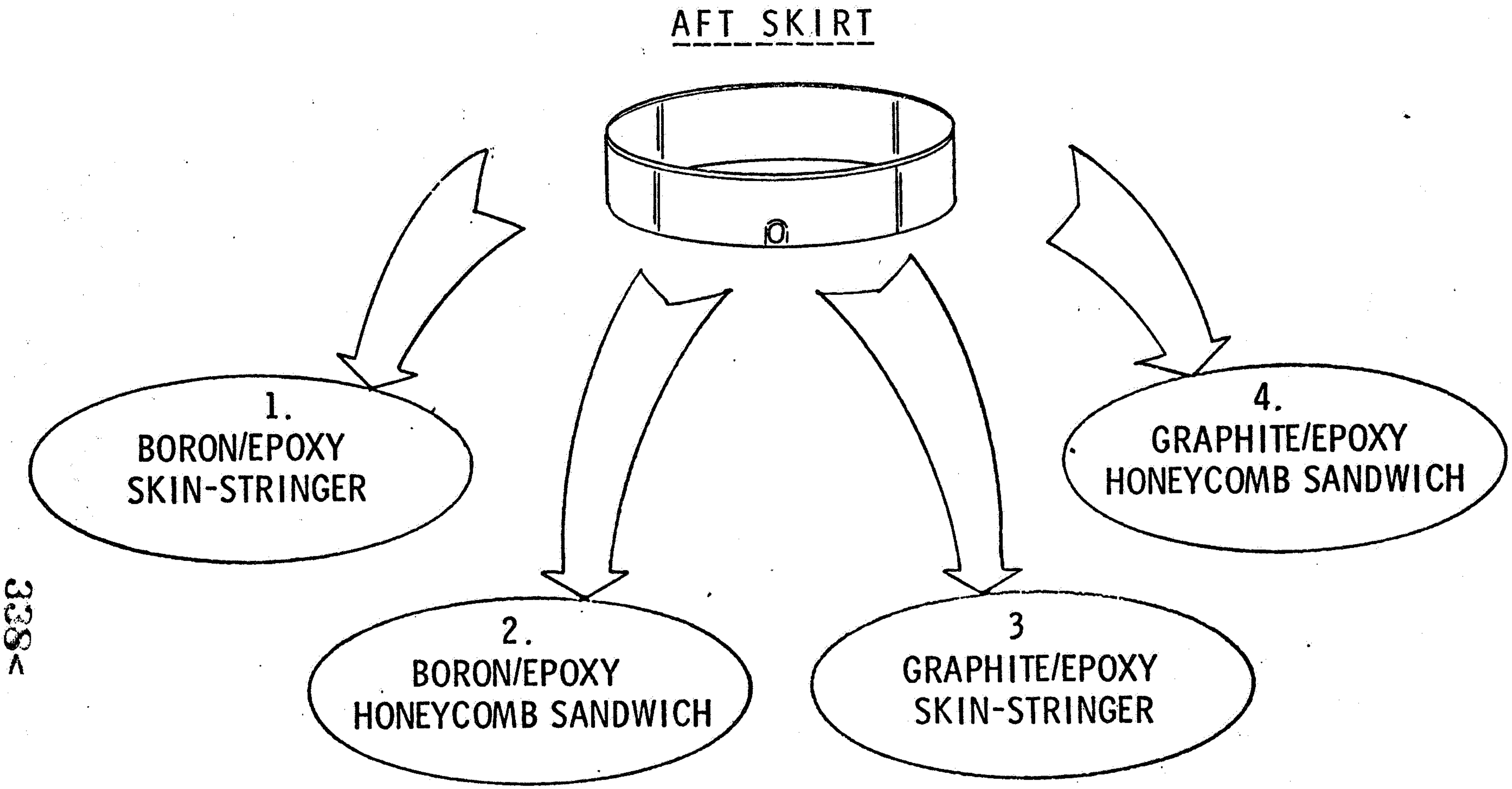

Figure 8. Aft Skirt Advanced Composite Structural Applications

338<

One of the composite application point designs, boron/epoxy skin-stringer, used the bi-composite joint concept. Composites of boron/epoxy are viable structural candidates, especially where stiffness to weight properties are desireable. Boron/epoxy has the advantage of a higher thermal coefficient of expansion than graphite/epoxy and is therefore more compatible for bonding to metal structures and also provides better thermal isolation properties than graphite/epoxy. Boron composites however, are more difficult to machine and the prepreg requires a 1/2 inch form radius and is difficult to drape. The boron/epoxy stringer reinforced skin concept given in Figure 9 is based on a laminate consisting of nine plies .007 thick with a laminate configuration of $[0/\pm 45/0/90/0/\mp 45/0]$. Titanium reinforcement is interleaved with the composite layers at the edges to provide for attachments.

The minimum weight boron/epoxy stringer has a dimension across the foot of 4.00 inches, a width across the crown of 2.09 inches and a height from crown to foot of 2.09 inches. The stringer spacing is 6.50 inches which results in 48 stringers per quarter panel. The boron/epoxy stringer has eight plies of composite with a laminate configuration of $[\pm 45_2]s$. The form radius is of course 1/2 inch.

A bi-composite joint was used in this study to counteract the 1/2 inch form radius at the stringer termination. Each end of the aft skirt boron/epoxy stringer interfaces with a three bolt attachment to adjacent stage hardware. One of the bolts is located on the stringer center line and one each on either side of the stringer through a bonded fitting as shown in Figure 10. A bi-composite transition was used to change from boron/epoxy to graphite/epoxy and the resulting 1/16 form radius. With the added layup flexibility a closeout type fitting could be designed into the end of the stringer. The smaller form radius also provided a more efficient bond for the adjacent molded chopped fiber fitting.

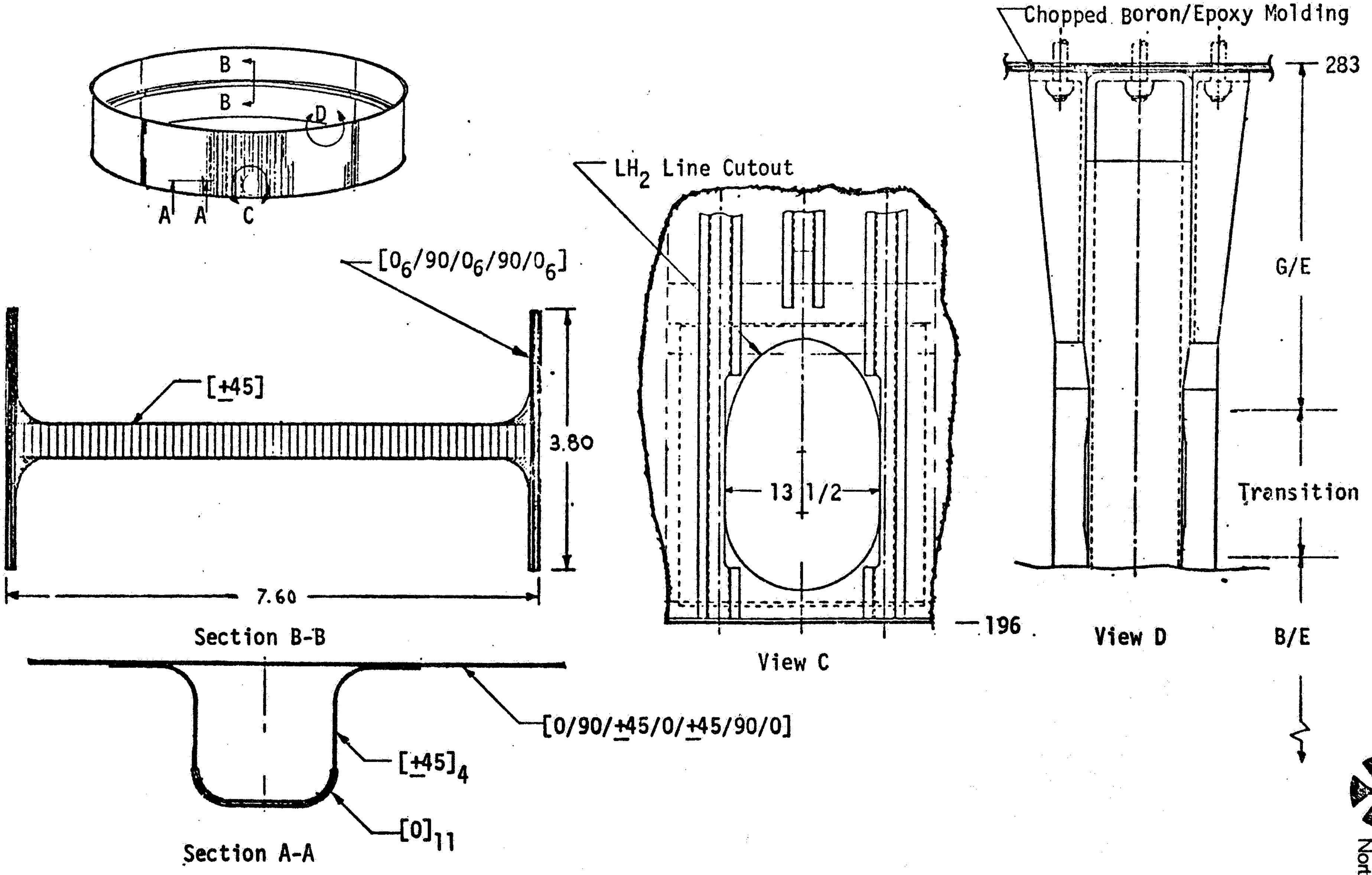

Figure 9. Boron/Epoxy Skirt With Bi-Composite Stringer

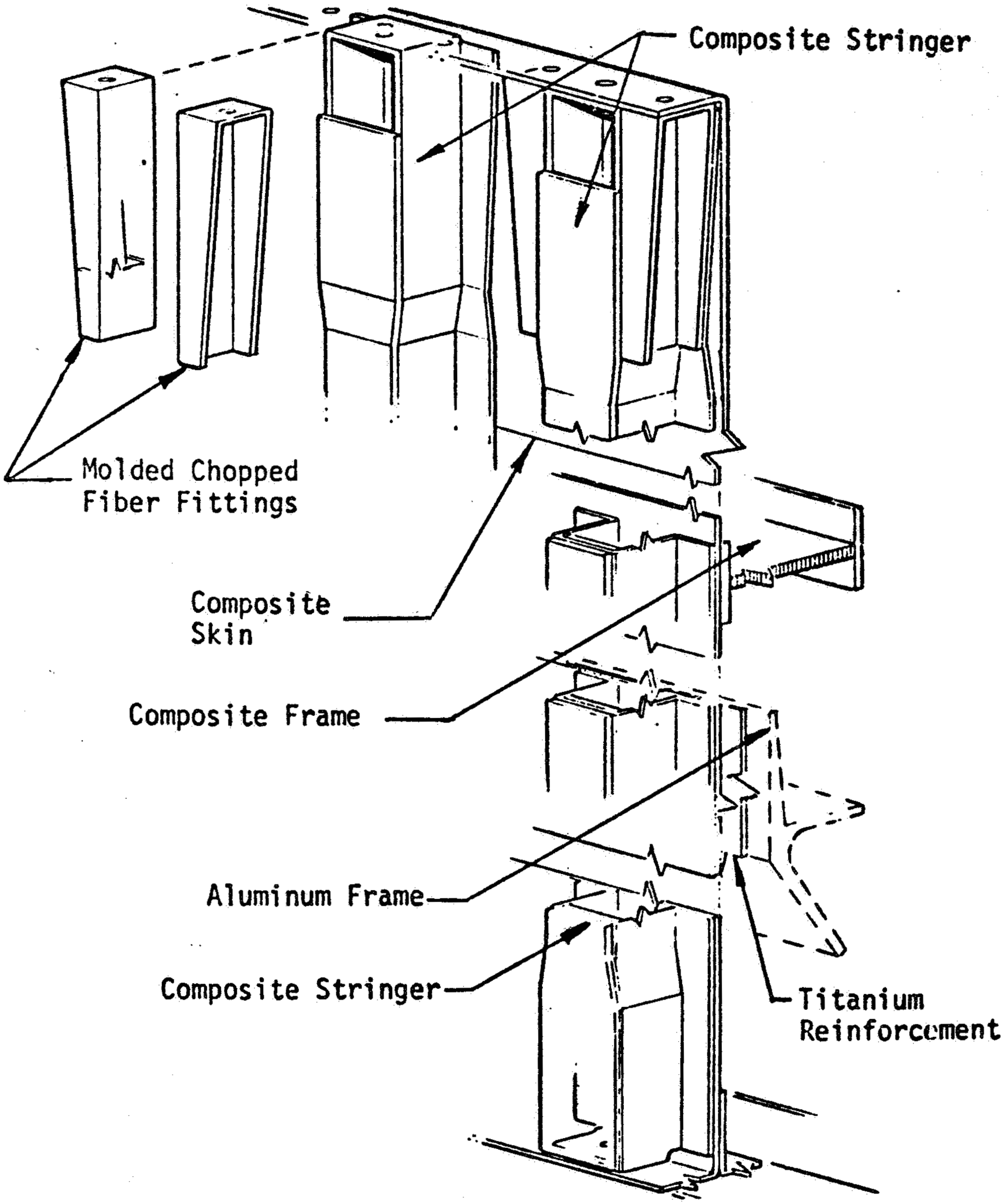

Figure 10. Boron/Epoxy Stringer

CONCLUSION

The bi-composite joint is applicable to a wide variety of design applications as illustrated in the preceeding discussion. It is an effective solution to many design problem areas such as thermal expansion and/or isolation and to satisfy requirements for varying structural load/strength requirements within a given structural component. It also provides an effective way to alleviate manufacturing producibility constraints such as forming radius and machining operations associated with the boron fiber materials.

The inherent flexibility in fabrication operations associated with the laminar ply lay-up technique of composite materials permits the inclusion of a bi-composite material transition section with very little impact on the fabrication operation. This fact in conjunction with the potential benefits to be gained make the bi-composite joint a very attractive concept for satisfying specific design requirements in composite material structures.

APPROVAL FOR
SCIENTIFIC AND TECHNICAL PRESENTATIONS AND PUBLICATIONS

TO : Jules M. Miller, M/S 105, 3251 DATE May 25, 1972

FROM : Aero-space Technologist, AGENCY CODE 113-32-01-21

SUBJECT: Material enclosed consist of ☐ Abstract ☐ Rough draft, L-______ ☒ Final paper for ☐ Article ☐ Paper ☒ Talk ☐ ______ entitled:

Small Scale Explosive Seam Welding

Authors (if not all Langley authors, indicate affiliation and describe nature of collaboration in par. 9):

L. J. Bement

1. Paper will be presented by: L. J. Bement Respective authors' tel. nos. (1) 827-3823 2 ______ 3, ______
2. ☒ Meeting ☐ Publication Name: Symposium on Welding, Bonding, and Fastening
3. Date and place of meeting: May 30-31, June 1, 1972 - Williamsburg, VA
4. Sponsor: NASA, George Washington University, & The American Society for Metals
5. Status of material: ☒ Data are in hand ☐ Data will be available by ______

   Classification of material: ☒ Unclassified ☐ Confidential ☐ Secret ☐ ______

   Previous approvals:

   ☐ None ☒ Rough draft From ☒ Langley ☒ Sponsor

   ☒ Abstract ☐ , ☒ Headquarters ☐

6. Previous or future publication(s) of essentially same material (if no NASA report will be published, state why):

Author(s)	Title	Status (If presented, state where, when)
L. J. Bement	Small Scale Explosive Welding of Aluminum	The Franklin Institute Philadelphia, PA Sept. 1971

7. Send present material to: Mr. Bland Stein
8. Deadline date of action: June 30, 1972
9. Additional comments (include statement regarding in-house review of paper for external publication):
10. ☒ References were checked by Technical Editing Section.
11. Name of technical reviewer or chairman of editing committee: ______ Tel. no. ______
12. Mr. Samos 3281, of Technology Utilization Office advised of new technology involved

    ☐ No new technology involved.

13. The undersigned, all divisions involved, concur in the above statements (sign names):

    ______ Section Head ______ Branch Head ______ Division Chief

    ______ Section Head ______ Branch Head ______ Division Chief

14. If "call for papers" or journal publication information is not in General Files, attach to indicate what is required.

    ☐ Copies of all correspondence received direct by authors regarding this matter were sent to Mr. Miller.

Submit original and one copy of this form with each action.

NASA Langley Form 96 (Dec 1970)

343<