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SUMMARY

As NASA enters the definition phase of the Space Station project, one of the important issues to be considered is structural material selection. The complexity of the Space Station and its long life requirement are two key factors which must be considered in the material selection process. Both aluminum and graphite/epoxy are considered as potential structural materials. This report presents advantages and disadvantages of these materials with respect to mechanical and thermal considerations, space environment, manufacturing, and cost.

INTRODUCTION

In January 1984, NASA was directed by President Reagan to develop a Space Station within a 10-year period. One of the first elements of this 10-year program is the definition phase during which the Space Station system as well as its many requirements will be defined. One of the myriad elements critical to the Space Station design is the selection of structural materials that are best suited for the subsystems and their applications. Before this can be done, the Space Station requirements and the structural material issues must be identified and understood. The purpose of this paper is to survey the structural material issues as they relate to the Space Station structural design.

For many years, aluminum has been a standard structural material in aircraft; however, advanced composite materials developed in recent years now make up a significant percentage of the structures of aircraft and spacecraft (refs. 1 and 2). The Space Shuttle orbiter, for example, utilizes composites in braces, struts, and most notably, the payload bay doors (ref. 2). As NASA develops designs for a Space Station, it is appropriate that recently matured materials such as advanced composites be considered along with more traditional materials such as aluminum. This paper examines the issues with respect to these two candidate classes of materials and focuses on structures such as module pressure shells and structural supports for large sections, including solar arrays and radiators.

SPACE STATION DESCRIPTION

The concept of a Space Station has been studied by NASA, the military, and aerospace industries since the 1960's. A general Space Station design has emerged from these studies. This general Space Station is a base in low Earth orbit (LEO) that has life support capability and an indefinite life. The design is modular in that modules or elements are attached to one another or to a base structure to form the Station configuration architecture. The Station includes its own power system and is periodically resupplied from Earth with essential consumables. The Space Station is evolutionary in that it can be characterized by growth of the overall system, especially with regard to power, crew size, and operational capability.

To facilitate the definition phase of the Space Station development, a reference configuration has been established (ref. 3). The concept is modular and solar powered. The elements of the Station would be delivered to orbit for assembly by six to eight Space Shuttle flights. The Station will support a crew of six with growth potential to support a crew of eighteen. The Station design will emphasize accommodation of user requirements for missions in science and applications, commercial ventures, and technology development. The power requirement is 75 kW for the initial Station and 300 kW for the growth Station. The Station will be resupplied at 90-day intervals with a logistics module that is carried in the cargo bay of the Space Shuttle orbiter (from the NASA Space Station Definition and Preliminary Design Request for Proposal, September 15, 1984). As the Station evolves to full operational capability, it will function as a staging base for vehicles being transferred to higher orbits.

The reference configuration is shown in figures 1 and 2. Figure 1 is an isometric view of the concept with the orbiter docked, and figure 2 shows front and side views. The reference configuration is gravity gradient stabilized; therefore, the pressurized modules having relatively larger masses will be nearer the Earth (at the bottom of the figures). Large truss structures above the modules support solar arrays, radiators, and payloads. A more detailed description of this concept may be found in reference 3.

STRUCTURAL DESIGN ISSUES

In this section, selected issues central to the material selection process for the Space Station are addressed. Specific structural design issues are categorized and discussed as follows: mechanical considerations, thermal considerations, space environment, manufacturing, and cost.

Mechanical Considerations

The Space Station structural designer is primarily interested in mechanical properties such as specific stiffness, specific strength, fracture, and fatigue of the candidate materials. Stiffness is the critical design issue for the large appendages on the Station, such as the solar arrays, the radiators, and the payload accommodation structure. The solar arrays will be one of the largest appendages on the Station, and as the power requirement for the Space Station increases, the size of the solar arrays increases. Initial guidelines have been set for the solar array fundamental frequency; the specified frequency range can be translated into stiffness guidelines. Specific strength, fracture, and fatigue are addressed in less detail, since guidelines for these material properties have yet to be defined.

<u>Specific stiffness</u>.- Various elements of the Space Station, such as solar arrays, thermal radiators, and the payload accommodation structure, may primarily be large flexible structures. It is anticipated that these structures typically may have a very low fundamental frequency, e.g., 0.008 to 0.012 Hz for solar array wings (ref. 4), which could pose an attitude control problem for the Space Station. Design requirements have not been established for all these structures, but in 1983, the NASA Space Station Concept Development Group set the fundamental frequency requirement for the solar arrays at >0.4 Hz. Once the fundamental frequency of the solar array was established, the support structure (strongback) for the arrays could be designed.

By designing a solar array strongback with high stiffness, the fundamental frequency requirement of >0.4 Hz can be achieved. As shown in table I, graphite/epoxy has a specific stiffness (E/ρ, where E is Young's modulus and ρ is the density) at least 3.6 times that of aluminum. One of the unique characteristics of composites is their ability to be tailored to meet a specific design requirement (ref. 5, p. 224). This property allows the designer to orient the fibers so as to take maximum advantage of the material properties. Current studies of solar array support structures (refs. 4 and 6) have considered a laminate construction of $(90/0_9)_S$, which has a longitudinal modulus of 275.8 GPa (40×10^6 lb/in^2). (See ref. 7, p. 116 for a description of the notation used to describe laminates.) A comparable aluminum structure (see table I) would weigh approximately 6.3 times the weight of the graphite/epoxy structure (private communication from Harold Bush, NASA Langley Research Center, Hampton, Virginia).

Space Station appendages such as solar arrays, radiators, and service structures are large because of the requirements for power, thermal control, and operations and servicing. Although an appendage such as a solar array support structure can be constructed of aluminum to meet the fundamental frequency requirement, prior studies have focused on structures fabricated from graphite/epoxy.

Specific strength, fracture, and fatigue.- Guidelines for specific strength, fracture, and fatigue have not been set; moreover, the Station is in the early design stages, and design loads for the structure are not defined. However, aluminum and graphite/epoxy can generally be compared with respect to these properties. Graphite/epoxy, which has a specific strength of 0.101 Mm (4.1×10^6 in.), compares favorably with aluminum, which has a specific strength of 0.020 Mm (0.8×10^6 in.) (ref. 8, p. 726).

The technology of fracture control for composites is not completely developed. The modes of failure for composites are more complex than those for aluminum, and consequently, technology for the two materials is not completely transferable. For example, linear fracture mechanics can be used when designing aluminum structures; however, composite materials, which are nonhomogeneous, require a more complex analysis.

The fatigue resistance of composite materials is superior to that of metals. Thus, composites may be considered a substitute for aluminum when fatigue is a major consideration.

Thermal Considerations

There are four material-related thermal considerations for the Space Station in its operating environment, LEO. The structural material issues of major concern in this area are the coefficient of thermal expansion (CTE) of the materials, microcracking in the structure, the conductivity of the materials, and the absorptance (α) and emittance (ε) of the surface materials.

Coefficient of thermal expansion.- The CTE of the materials is considered because the structure will be thermally cycled in its operating environment. In orbit, the Space Station structure will be exposed to constant temperature cycling over a long time period as the Station moves in and out of the Earth's shadow. Thermal cycling over the projected lifetime of the Space Station may lead to thermal fatigue problems in the structure.

As temperatures of the Space Station change, a structure made with aluminum, which has a high CTE, will expand and contract substantially. This expansion and contraction becomes more critical with larger structures. A large truss structure with thermally induced stresses may bind up and cause problems if a truss element needed to be replaced. Also, structures with precision shape and alignment requirements have small tolerances for change in the thermal environment.

The CTE values for 2219 aluminum alloy range from 20.8×10^{-6}/K to 24.4×10^{-6}/K (11.5×10^{-6}/°F to 13.6×10^{-6}/°F), depending on temperature (ref. 9). The CTE of aluminum is relatively high, and the Space Station structure designer must be aware of the potential problems in using this material, especially in large structures. As mentioned earlier, the CTE of a graphite/epoxy structure can be tailored by the designer because the laminae have different expansion characteristics. References 10 through 13 present data on CTE values for graphite/epoxy under varying conditions. These values range from -0.86×10^{-6}/K to 2.3×10^{-6}/K (-0.48×10^{-6}/°F to 1.3×10^{-6}/°F), depending on many factors, including the material system, layup, and temperature. Since graphite/epoxy can be tailored to meet a specific design requirement, structures with a low or even zero CTE can be produced. Use of these materials should minimize and perhaps eliminate the expansion problems. According to a study on microcracking effects (ref. 14), the CTE of a composite material is affected by microcracking of the material.

The thermal cycling of the Space Station structure is a concern because over the lifetime of the Station, the structure will be subject to thermal fatigue. This thermal fatigue must be controlled to meet the Station's indefinite lifetime requirement. In a study performed to characterize the thermomechanical behavior of composite tubes (ref. 15), the effect of thermal cycling on CTE values was investigated. Composite tubes 1 m (3.27 ft) in length were thermally cycled between -150°C (-238°F) and 70°C (158°F), and CTE measurements were taken. The results of these measurements, shown in figure 3, indicate more uniform CTE values after thermal cycling.

Microcracking.- Because of the large difference in the CTE values for the fiber and for the resin, composite materials have a tendency for microcracks to develop as a result of thermal cycling. Microcracks are tiny cracks in the composite matrix which run parallel to the direction of the fiber. Microcracking is unique to composite materials and can cause changes in the CTE characteristics in the short term and delamination in the long term. In a study of thermomechanical behavior (ref. 15), graphite/epoxy tubes were fabricated and subjected to slow thermal cycling from -150°C (-238°F) to 80°C (176°F) in vacuum. Test results indicate that (1) continued thermal cycling from 0 to 2000 cycles increases the number of microcracks, (2) the speed of cycling has a negligible effect on microcrack generation, and (3) at room temperature, the CTE decreases as the number of microcracks increases.

Conductivity.- The need to maintain a narrow range of temperature on the Space Station to accommodate requirements for equipment and life support will make thermal control for the Space Station a critical element in the design process. The use of structural materials with good thermal conductivity properties is therefore desirable. The room temperature thermal conductivity for 2219 aluminum alloy is 124.6 W/m-K (72 Btu-ft/hr-ft^2-°F) (ref. 16). The conductivity for graphite/epoxy is 0.50 to 0.62 W/m-K (0.29 to 0.36 Btu-ft/hr-ft^2-°F). Table II shows the conductivity values at various temperatures for this system (private communication from Al Vicario, Hercules, Inc., Magna, Utah).

Absorptance and emittance.- The absorptance and emittance of a material are important in an environment in which the solar insolations and other heat source temperatures are constantly changing as are the temperatures of the materials that must be controlled within certain bounds. The radiative requirement of a material is related to the function of the structure. For example, structures with no internal heat require a high absorptance and a low emittance, whereas structures with internal heat, such as the modules, require a high absorptance and a high emittance. Radiators, used to dissipate heat, will require a low absorptance and a very high emittance.

The absorptance values for aluminum vary from 0.0 to 0.4, depending on composition and heat treatment (ref. 17). The emittance values vary from 0.0 to 0.4, depending on surface finish (e.g., polished or roughened) (ref. 17). Both emittance and absorptance values for graphite/epoxy are approximately 0.9.

It should be noted that protective thermal coatings can and probably will be used to enhance the thermal control performance of the structure in the Space Station environment. By controlling the temperature of the structure through the use of thermal coatings which change the absorptance and emittance values of the structure surface, many thermal problems can be partially or completely eliminated. The degree to which the structural material is susceptible to the thermal environment will be a major factor in the selection of the coating. (Coatings are further addressed in the section "Space Environment.")

Space Environment

The Space Station is expected to operate in the LEO environment for an indefinite period of time. Long life of the structure will depend, in part, on the capability of the structural materials to withstand the potential hazards in this environment. This section addresses the effects of outgassing and contamination, chemical inertness, radiation, and impact on the Space Station.

Outgassing.- In the Station environment, there are several sources of contamination, including exhausts from thrusters, surface outgassing, leaks, and dumps. Optical sensors and thermal control surfaces are examples of equipment that is particularly sensitive to contamination. The structural material issue here concerns the material as a contamination source via surface outgassing and toxic fumes.

Structural metals such as aluminum lose material by evaporation in a low-pressure environment. Calculations show that aluminum loses 0.102 cm/yr (0.040 in/yr) in a vacuum at 810°C (1490°F) (ref. 18). This temperature is beyond the operating range of aluminum, and the material loss is considered insignificant.

The outgassing of composite materials involves the loss of material volatiles in a vacuum. These volatiles include absorbed gases, solvents, and decomposition products (ref. 19). In a compilation of outgassing data (ref. 20), the total mass loss (TML) values for graphite/epoxy range from 0.11 to 0.58 percent, and the collected volatile condensable materials (CVCM) values for graphite/epoxy range from 0.00 to 0.01 percent.

Atomic oxygen.- On recent flights of the Space Shuttle orbiter, experiments have been conducted to investigate the degradation of certain materials as a result of exposure to atomic oxygen, which is predominant in the LEO environment. The extent of the problem is still under investigation.

Initially, atomic oxygen, because of its high chemical reactivity, was found to cause significant weight loss in polyimide film, which is used as a protective coating for solar arrays. The phenomenon is not limited to the power subsystem but extends to all elements of the Space Station constructed from materials that are susceptible to atomic oxygen degradation. In a Space Transportation System (STS-8) experiment, graphite/epoxy specimens were flown in an orientation normal to the velocity vector at a fluence of 3.5×10^{20} atoms/cm^2 for 41.75 hours. Data from the experiment indicate a typical material loss of approximately 9.1 μm (0.36 mil). The material loss appears to be independent of material thickness. Samples which had been coated showed no material loss after the flight. The coatings used on the graphite/epoxy were (1) a second surface mirror coating, which is heavy, delicate, and difficult to clean and (2) a sputtered metallic coating, still under development, which is durable and easy to clean. The flight test results indicate that the atomic oxygen degradation of graphite/epoxy can be controlled with the use of either of these protective coatings. Aluminum samples were also flown on the orbiter but did not experience any measurable weight loss during the flight (ref. 21).

Fireworthiness.- Fireworthiness of materials for aircraft and spacecraft is a concern when using materials which will burn and give off toxic fumes while burning. Recent studies in this area have dealt with materials as used in aircraft structures; however, the results are appropriate for spacecraft applications as well. In a recent study, the issue of fireworthy composites for aircraft is addressed (ref. 22). Figure 4 is a plot of data from this study and shows the fire endurance of aluminum and graphite/epoxy. Graphite/epoxy has a lower time to structural collapse (thermochemical failure) than aluminum. The challenge is to improve the fire resistance of the composite material without significant loss of material properties or increasingly complex processing methods. The authors of reference 22 looked at a number of ways to improve the fireworthiness of both interior and primary structures and found that bismaleimide composites used in interior structures and halogen-modified rubber-toughened composites used in primary structures have a satisfactory combination of properties for fireworthiness considerations. A study was conducted to improve fire resistance characteristics of resin systems for interior materials (ref. 23). It concluded that fire resistance characteristics could be improved by replacing epoxy with modified phenolic resins while maintaining acceptable mechanical properties.

Radiation.- Materials on the Space Station will be exposed to both high-energy electromagnetic radiation and particulate emissions in the LEO environment. A primary source of electromagnetic radiation is the Sun. The electromagnetic wavelength regions include low-energy gamma rays, x-rays, ultraviolet light, visible light, infrared light, and microwaves. There are several sources of particulate radiation, including solar cosmic radiation, galactic cosmic radiation, and LEO-trapped radiation. Solar cosmic radiation results from solar activity, which is cyclic in nature and is characterized by solar flares. The primary constituent of this radiation consists of protons with energies ranging from 1 to 100 MeV and greater. Galactic cosmic radiation consists primarily of high-energy protons. A small percentage of the cosmic rays consists of nuclei with high atomic numbers (heavy ions). Finally, LEO-trapped radiation consists of high-energy electrons and protons (refs. 24, 25, and 26).

Damage to spacecraft materials from these radiation sources occurs primarily through ionization. For aluminum in LEO, there is no significant damage caused by ionization (ref. 24). Changes in material strength, ductility, and creep rate in aluminum materials as a result of exposure to radiation occur at a dosage many times higher than the material would receive in LEO (refs. 27 and 28). However, the properties of some materials can be degraded in this environment. Ionization of composite materials can result in the breakdown of chemical bonds (ref. 24). This chemical breakdown can lead to the destruction of surface layers of the matrix material. In a study on the effects of the space environment on graphite/epoxy (ref. 29), various properties of the material were investigated after exposure to gamma radiation. The highest dosage level was equivalent to a 3-year orbit life. Results of the study indicate that glass transition temperatures decrease after exposure (see table III) and that interlaminar shear strength initially increases and then decreases. (See fig. 5.) In another study on the effects of radiation of graphite fiber composites (ref. 30), samples were exposed to electron radiation and subsequently tested in flexure and interlaminar shear. The specimens were irradiated with 0.5-MeV electrons to a maximum dosage of 10 000 Mrads. The study data indicate that flexural strength properties were not significantly affected by radiation and that after an initial increase, the interlaminar shear strength decreased with increasing radiation.

Because of the relatively low radiation dosage levels in LEO, aluminum is not significantly damaged by radiation in this environment. However, ionization can damage or destroy the surface layers of the matrix material of a graphite/epoxy composite and cause the degradation of surface properties of the material.

Impact.- Both micrometeoroids and space debris in LEO represent a potential hazard to orbiting spacecraft, especially the Space Station, which will have a large area and a long lifetime. Approximately 4500 objects are tracked by the North American Aerospace Defense Command (NORAD) (ref. 31), and it is estimated that there are numerous objects less than 4 cm (1.6 in.) in diameter that cannot be detected by the NORAD sensors. Space debris in LEO travels at approximately 8 km/s (26 247 ft/sec), and micrometeoroids travel at approximately 20 km/s (65 617 ft/sec); these objects have the potential to severely damage spacecraft. As stated in an AIAA position paper on space debris: "A 1-cm object in low Earth orbit could penetrate a 5-cm thickness of solid aluminum" (ref. 32). The probability that an object tracked by NORAD will collide with a Space Station 100 m (328 ft) in diameter with a 10-year lifetime is 0.1 (ref. 33). When untracked objects are also considered, the potential for impact increases. These collisions could occur anywhere on the Space Station structure; however, damage to the pressurized modules could cause leakage of the internal atmosphere and threaten the lives and safety of the crew. Therefore, this potential damage is of particular concern.

Studies have been conducted to evaluate the damage to structures caused by the high-velocity impact of a projectile. In a study designed to validate threshold penetration relations (private communication from Emilio Alfaro-Bou, NASA Langley Research Center, Hampton, Virginia), a 0.64-cm (0.25-in.) thick section of aluminum was impacted at a velocity of 9 754 m/s (32 000 ft/sec), a velocity near the average velocity of space debris. (The actual wall thickness of a module will probably be less than 0.64 cm (0.25 in.).) Figure 6 shows the damaged aluminum section which has been completely penetrated. Available data on high-velocity impact of graphite/epoxy are very limited to date. NASA Johnson Space Center is evaluating the effects of hypervelocity impact on graphite/epoxy, but the study is still in progress, and results have not been published. Graphite/epoxy materials have been impacted at velocities ranging from 4 km/s (13 123 ft/sec) to 8 km/s (26 247 ft/sec). Preliminary results indicate that there is surface damage including peeling and delamination, but internal damage is difficult to assess. Preliminary C-scan results indicate that internal damage may be more extensive than surface damage (private communication from Jeanne Lee Crews, NASA Johnson Space Center, Houston, Texas). The use of standoff bumper shields is being evaluated as a means of protection for the pressure vessel skin.

Low-velocity impact is also a potential problem for all elements of the Station. This type of impact can occur in handling of the Space Station both on the ground during manufacturing and preparation for launch and in space during deployment, assembly, and repair of elements. Low-velocity-impact studies for Space Station structures are important when considering handling of Station elements. Generally, the energy levels for low-velocity impact are simulated by the impact of a ball with a specific diameter.

A study was conducted to evaluate erosion of metal surfaces (ref. 34). Part of the testing included the impact of 6061 aluminum alloy with a 3.2-mm (0.126-in.) diameter steel ball at 140 m/s (459 ft/sec). The aluminum surface showed deformation and fracture. In another study (ref. 35), various metals were impacted by 0.476-cm (0.1875-in.) diameter ball bearings at relatively low velocities, and results show that impacts on 7075 aluminum alloy at 143 m/s (469 ft/sec) formed craters in the material.

Several studies of low-velocity impact of graphite/epoxy have been conducted. In a 1979 study (ref. 36), specimens were impacted by 1.27-cm (0.5-in.) diameter aluminum projectiles at velocities from 30 m/s (98 ft/sec) to 140 m/s (459 ft/sec). Figure 7 is a photograph showing interior damage in two laminates impacted at different velocities. Both laminates show a similar damage pattern of cracking in a v-shape. Laminates had similar surface damage patterns which were more severe on the back surface than on the contact surface. For laminates impacted at the higher velocities of the range mentioned above, there were surface cracks and delamination on the back surface and a shallow depression on the contact surface. The occurrence of more extensive internal damage at these velocities is similar to the internal damage of composite specimens impacted at very high velocities.

In summarizing the space environmental effects on the materials, the primary concerns in the use of an aluminum Space Station structure are the potential damage that can be caused by the impact of micrometeoroids and/or space debris and the potential for damage in handling. However, there is no indication that aluminum is susceptible to problems caused by outgassing/contamination, atomic oxygen, or radiation in LEO. On the other hand, it is apparent that a graphite/epoxy structure is susceptible to all the hazards listed in this section. Again, it should be noted that solutions to such environmental problems are being considered and developed. For example, through the use of protective coatings and bumper shields, the susceptibility of the materials to the hazards of atomic oxygen and micrometeoroid impact, respectively, may be reduced to an acceptable level.

Manufacturing

The manufacturing techniques used for the construction of a Space Station element will vary extensively and depend on the material used in construction. The efficiency and cost of these techniques will play a significant role in determining what structural materials should be used.

The Space Station module or pressure shell design will be based on proven designs for pressure vessels. These designs include (1) a monocoque, or a single-skin design, (2) a semimonocoque design in which the outer skin is stiffened, usually with ring frames and/or skin stringers, (3) a sandwich type of construction, and (4) an isogrid design. The manufacturing techniques discussed in this report are based on one or more of these designs.

A white paper on the Space Station module was recently published (ref. 37). The material specified for the module construction is 2219-T851 aluminum alloy. The cylinder structure is an integrally stiffened skin with ring frames. The technique considered for manufacture of such a module is a machined 5.1-cm (2-in.) thick plate, formed and welded. The machined skin thickness is 0.18 cm (0.07 in.). The 2219 aluminum alloy is chosen in part because it has excellent weldability. An analysis of this design showed that this structure could withstand predicted pressure loads and Shuttle flight loads.

Three composite module designs - (1) a single-skin structure, (2) a sandwich structure, and (3) a ring and stringer structure - have been evaluated. Each of the concepts would be manufactured by a filament-winding process. These three designs were compared with a baseline aluminum design (a stiffened structure using ring frames and longitudinal stiffeners). All graphite/epoxy designs had lower weight, a greater minimum factor-of-safety, greater micrometeoroid protection, and a lower relative cost factor than the aluminum baseline, except that the single-skin structure weighed slightly more than the baseline design (private communication from Al Vicario, Hercules, Inc., Magna, Utah).

Another type of design for pressure shells is an isogrid structure, a lattice of rib members in a pattern of contiguous isosceles triangles. A skin stiffened by an isogrid structure usually has good tension, compression, and shear strengths. This type of structure was originally applied to the floors and walls of the orbital workshop module of Skylab to provide adaptable attachment points for mounting equipment (refs. 38 and 39). In a study on isogrid configurations which have been used in launch vehicles, propellant tanks, and Skylab floors and walls (ref. 38), an isogrid cylindrical adapter was fabricated from 2024-T851 aluminum panels and used to replace an aluminum skin-stringer, frame-stabilized adapter. The isogrid structure showed advantages in weight and strength and had fewer parts, which resulted in a manufacturing cost reduction of 40 percent. This study also looked at the application of isogrid construction to the Space Tug, a reusable upper stage designed to extend the range of the orbiter. The study showed that sections for the Tug shell could be fabricated from graphite/epoxy at a low cost by using thermal pressure-forming methods.

Support structures for large sections, such as solar arrays and radiators, will likely be truss structures. A fabrication technique for the tapered nestable tubes, which are the typical elements for the truss structure described in the stiffness section, has been defined (ref. 40). The tubes are filament wound on a heated mandrel, and the tube and mandrel assembly are overwrapped with a bag and cured. No work with aluminum is being performed in this area.

The techniques used in the manufacture of a structure depend largely on the type of material used and the type of structure. The need for a specific type of structure, for example, a structure that will provide a flexible arrangement of attachment points for mounting equipment, may have a strong influence on the manufacturing techniques used. Likewise, the type of material used, graphite/epoxy or aluminum, will dictate manufacturing techniques. Advantages of one process over another are dependent on many factors, such as cost of materials, whether the process is labor intensive or capital intensive, the number of units to be produced, and the number of parts needed for each unit. The techniques for machining composites to achieve the same quality that is standard with aluminum require more effort and are more time consuming. For example, the machining of some composites requires slow drilling speeds and diamond-impregnated tools. The efficiency and cost effectiveness in manufacture will be key factors in the selection of structural materials for the Space Station.

Cost

Direct cost issues related to structural materials for the Space Station are materials costs and manufacturing costs. Indirect cost issues involve such items as Space Station weight, propulsion, attitude control, and protective surface coatings, as well as nonhardware items, such as engineering, testing, servicing/replacement, and operations. At the time, it appears premature to address issues regarding the indirect cost items except, perhaps, weight. This report compares aluminum and graphite/epoxy as candidate materials. To estimate the cost of the Space Station structure utilizing either of these materials is beyond the scope of this paper. Therefore, this section will look at cost comparisons based on past experience. Direct costs of both materials and manufacturing are addressed. Indirect costs are addressed only by examining the effects of material selection on the weight of the Space Station.

Although the use of composite materials in aircraft and spacecraft has been shown to be advantageous over the use of aluminum with respect to weight savings, there has been a disadvantage with the high cost of the materials and of their early manufacturing techniques. As composite materials have developed, however, the cost has decreased because of greater use of the material in a production mode, a reduction in the parts needed, and improved manufacturing techniques. Current price quotes for aluminum and graphite/epoxy materials (based on a 10 000-lb order) are as follows: $2.67/lb for standard 5.1-cm (2-in.) thick aluminum sheet and $50/lb for graphite/epoxy prepreg for filament winding. Without evaluating a specific structure and specific manufacturing process, it is difficult to assess whether this cost difference will change when manufacturing processes are considered.

Historically, in the aircraft industry, there has been a 20- to 25-percent reduction in cost in replacement of aluminum structures with composites. In a study on the application of composites to the B-1 vertical stabilizer (ref. 41), the torque box was redesigned by using composite materials. The new design has a material distribution by weight of 56 percent graphite/epoxy, 39 percent metal and 5 percent other. The composite torque box has a reduced part count, an 18-percent weight savings, and a 12.9-percent cost savings over the baseline aluminum design. A study on an advanced composite fuselage structure (ref. 42) involved the use of composites in the fuselage of the Northrop YF-17 fighter aircraft. The baseline design has an aluminum longeron and frame with a stressed-skin construction, and the composite design has a honeycomb sandwich panel construction with graphite/epoxy face sheets and an aluminum honeycomb core. The composite design has a 30-percent cost savings and a 41-percent weight savings over the baseline aluminum design. Finally, in a study conducted for the Aircraft Energy Efficiency (ACEE) Project (ref. 43), the Boeing 737 horizontal stabilizer was fabricated from graphite/epoxy. Based on the assumptions of (1) a five-shipset production, (2) use of advanced manufacturing techniques, and (3) lower cost per pound for material as usage increases, the study concludes that the cost of a graphite/epoxy stabilizer will be comparable to that of a metal stabilizer.

Experience has shown that composites are becoming more cost competitive with aluminum materials. It is difficult to make a direct comparison of the cost of structures fabricated with these two materials, since the structures and manufacturing techniques may vary according to the material used.

CONCLUDING REMARKS

As the development of a low Earth orbit (LEO) Space Station begins, designers are confronted with the task of designing a highly complex structure with a very long lifetime. In the Space Station structure, there are many issues which require attention to meet this task. This report has examined structural material issues in the following areas: mechanical considerations, thermal considerations, space environment, manufacturing, and cost. Each issue has been examined to determine its relationship to the Station and its operations and to evaluate results of past and present research in the area.

Aluminum and graphite/epoxy have been considered in this process. Both materials have properties which may be particularly advantageous for Space Station applications for different reasons. For example, aluminum is resistant to degradation caused by radiation and atomic oxygen and does not outgas in LEO. Graphite/epoxy can be designed to have high stiffness and a tailorable coefficient of thermal expansion, which may be important for structures which experience severe temperature changes. However, both materials are susceptible to damage in the space environment (e.g., damage from the impact of micrometeoroids and/or space debris) and may require some form of protection to withstand the environment for long periods of time.

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TABLE I.- SPECIFIC STIFFNESS OF ALUMINUM AND GRAPHITE/EPOXY

Material	Specific stiffness, E/ρ	
	Mm	in.
2219 aluminum	2.6	1.0×10^8
Low-modulus graphite/epoxy	[a]9.1	[a]3.6
High-modulus graphite/epoxy	[a]16.0	[a]6.3

[a]Properties measured parallel to fiber direction.

TABLE II.- CONDUCTIVITY VALUES FOR GRAPHITE/EPOXY

Temperature		Conductivity	
K	°F	W/m-K	Btu-ft/hr-ft^2-°F
214	-75	0.36 - 0.45	0.21 - 0.26
255	0	0.45 - 0.55	0.26 - 0.32
297	75	0.50 - 0.62	0.29 - 0.36
366	200	0.59 - 0.69	0.34 - 0.4
422	300	0.59 - 0.69	0.34 - 0.4

TABLE III.- GLASS TRANSITION TEMPERATURES IN IRRADIATED GRAPHITE/EPOXY

[From ref. 29]

Total dosage, rads	Glass transition temperature, °C			
	Room temperature; open to atmosphere	Room temperature; vacuum	100°C; open to atmosphere	100°C; vacuum
0	260			
4.4×10^7	231	230	236	235
8.8×10^7	230	230	239	230
1.4×10^8	234	235	240	238
3.2×10^8	225		238	237

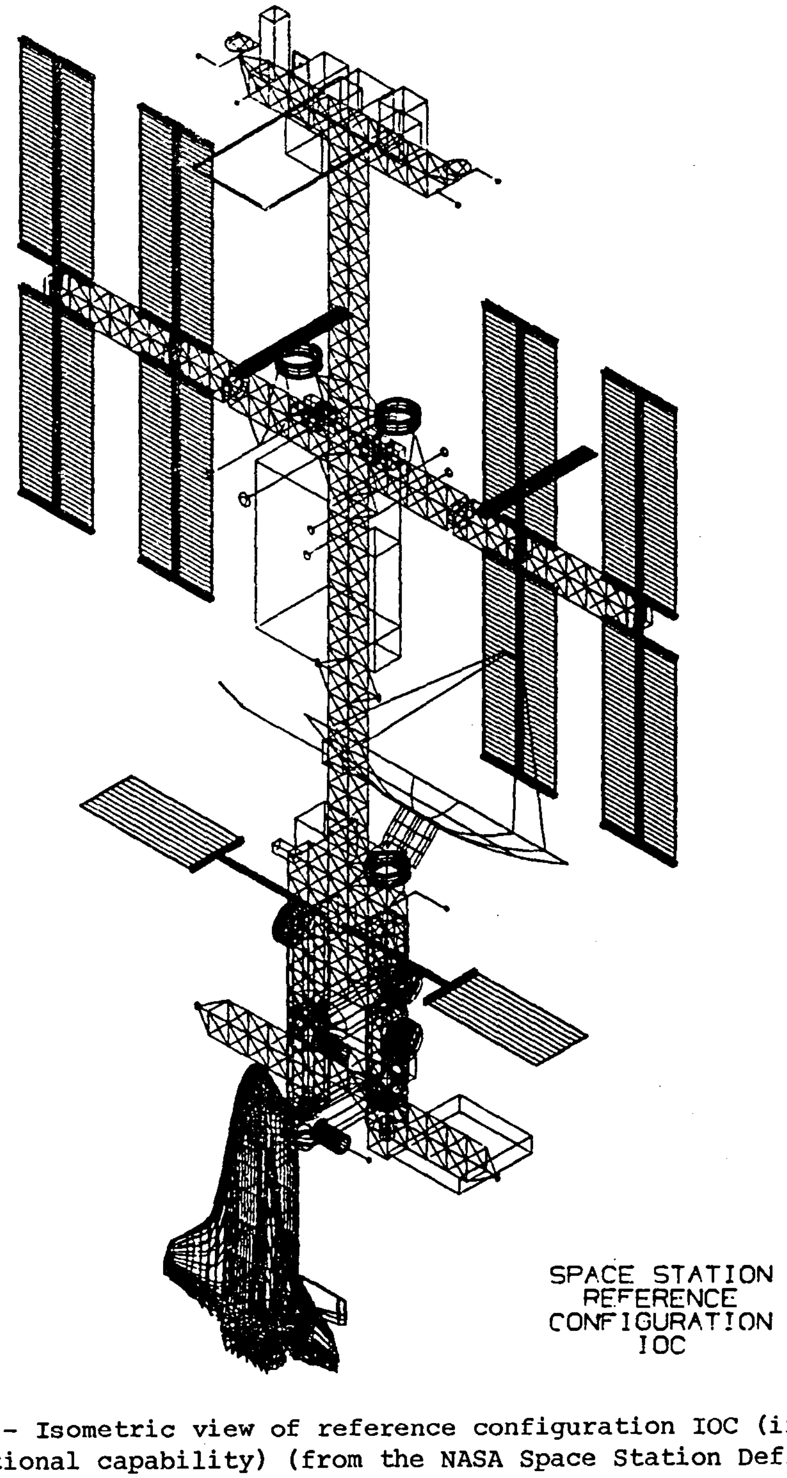

Figure 1.- Isometric view of reference configuration IOC (initial operational capability) (from the NASA Space Station Definition and Preliminary Design Request for Proposal, September 15, 1984).

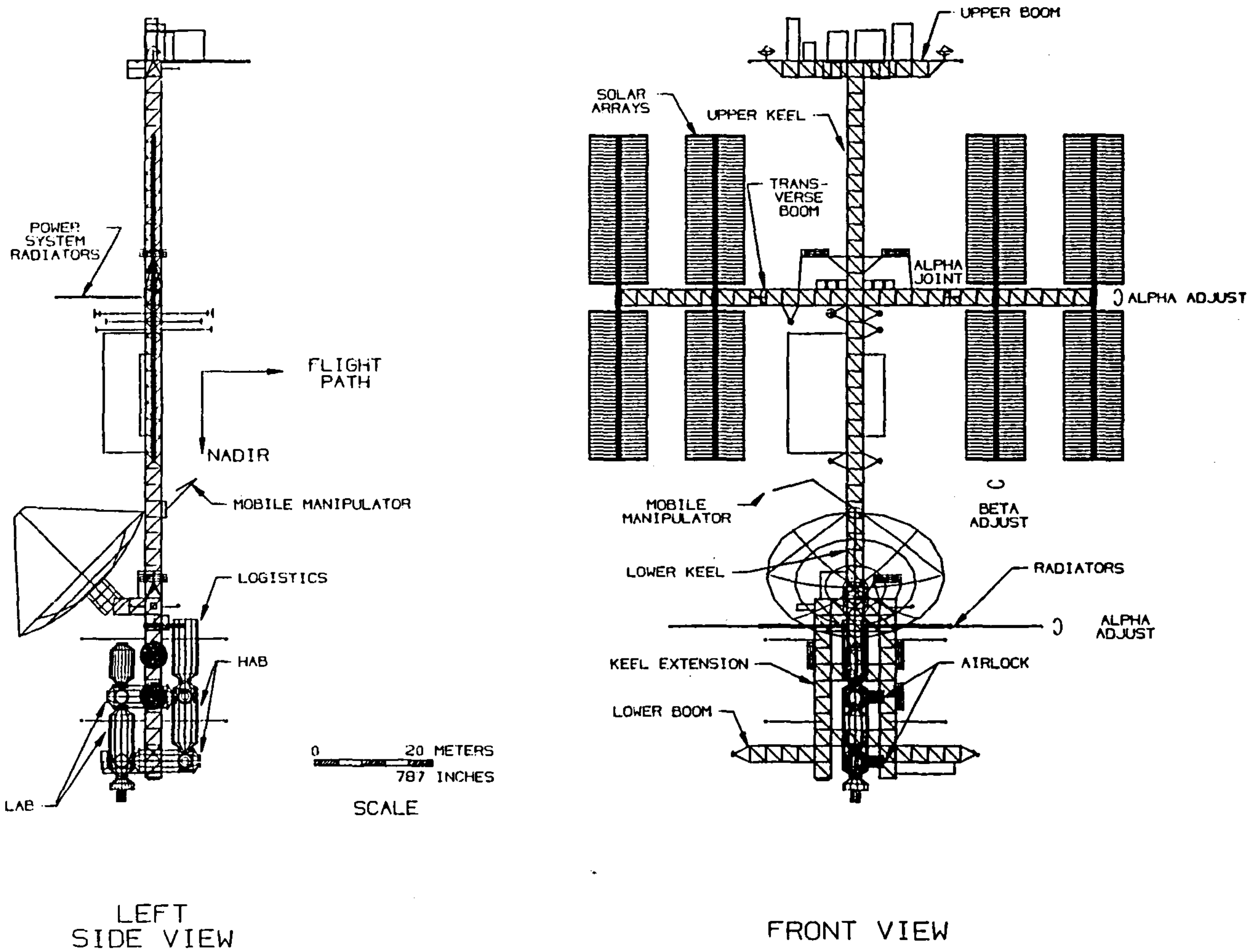

Figure 2.- Front and side views of reference configuration (from the NASA Space Station Definition and Preliminary Design Request for Proposal, September 15, 1984). HAB denotes habitation module.

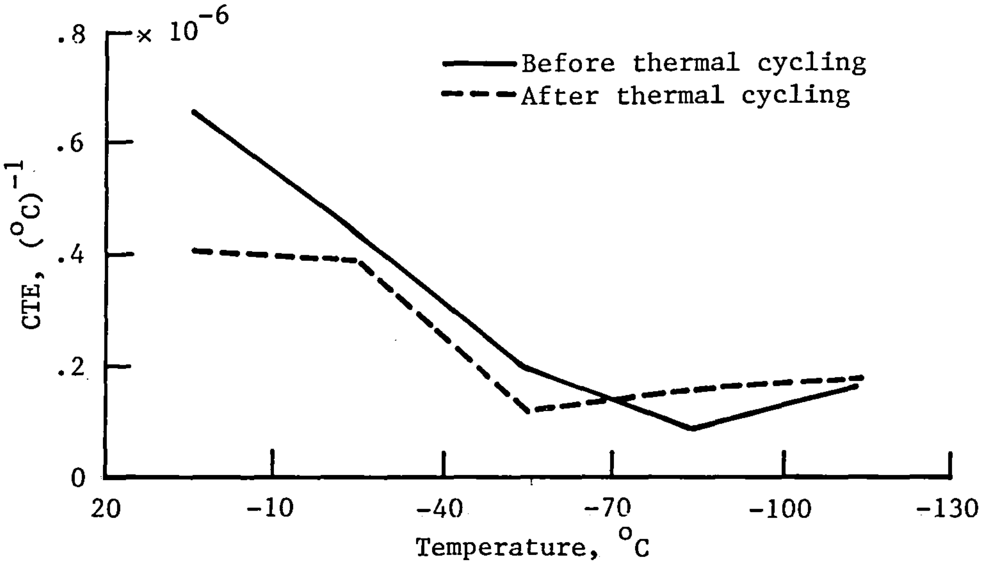

Figure 3.- Influence of thermal cycling on the CTE of composite tubes (data from ref. 15).

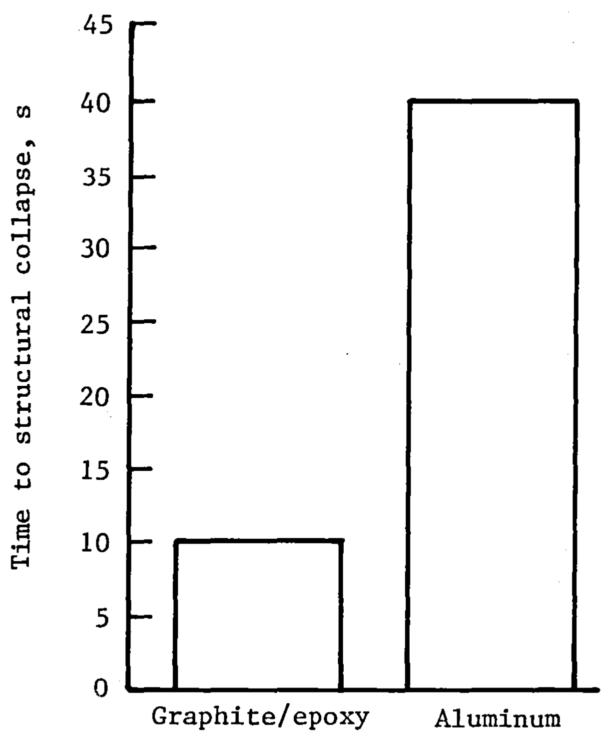

Figure 4.- Fire endurance of aluminum and graphite/epoxy (data from ref. 22).

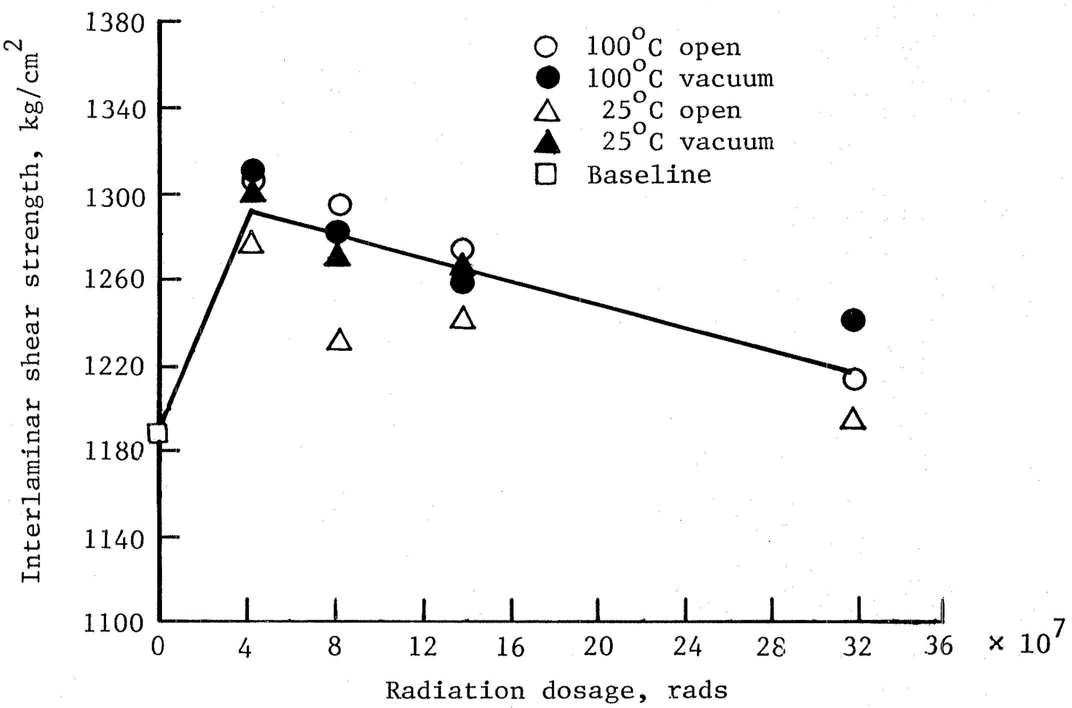

Figure 5.- Interlaminar shear strength of composites irradiated with gamma radiation (data from ref. 29).

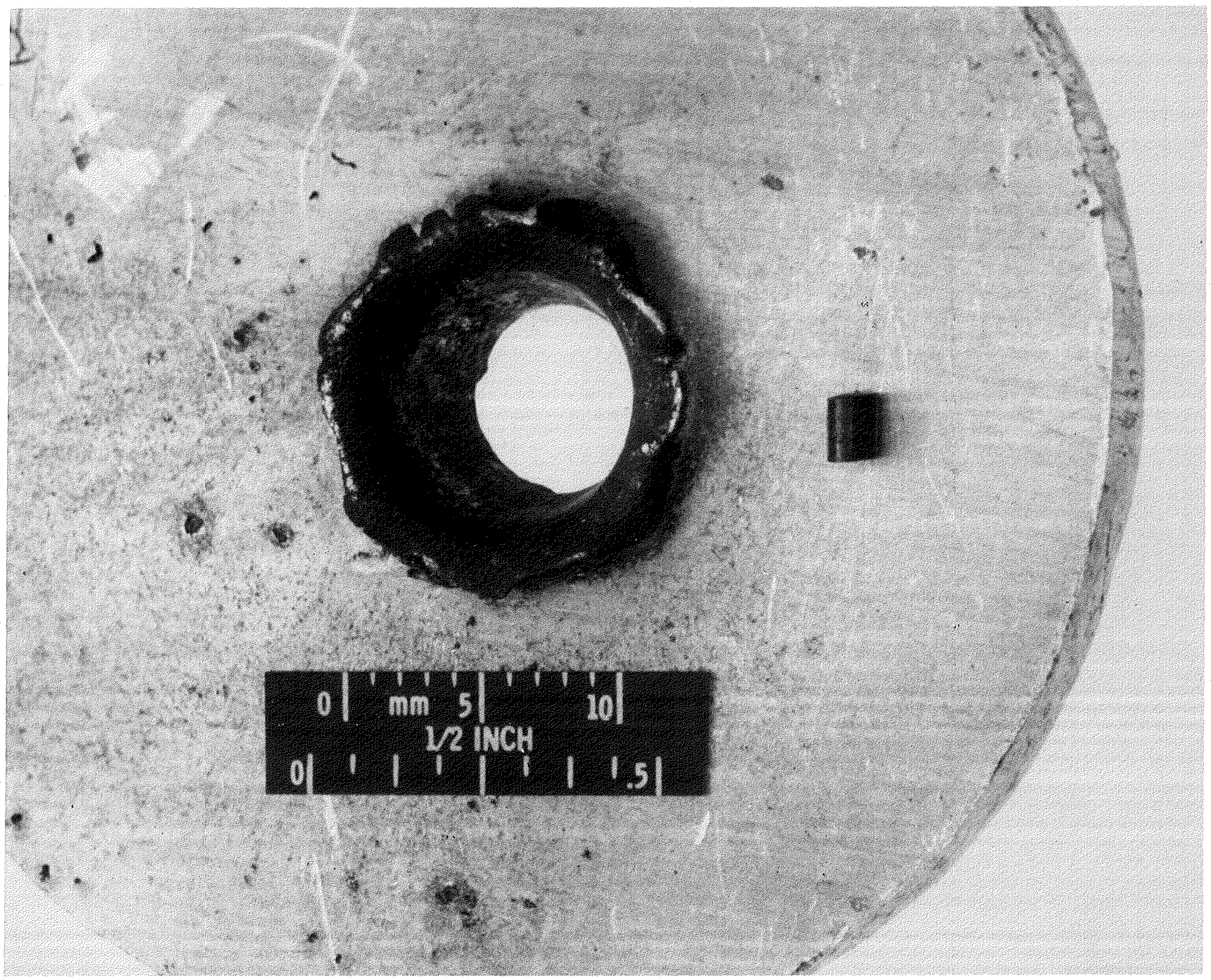

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Figure 6.- Aluminum section impacted at 9 754 m/s (32 000 ft/sec).

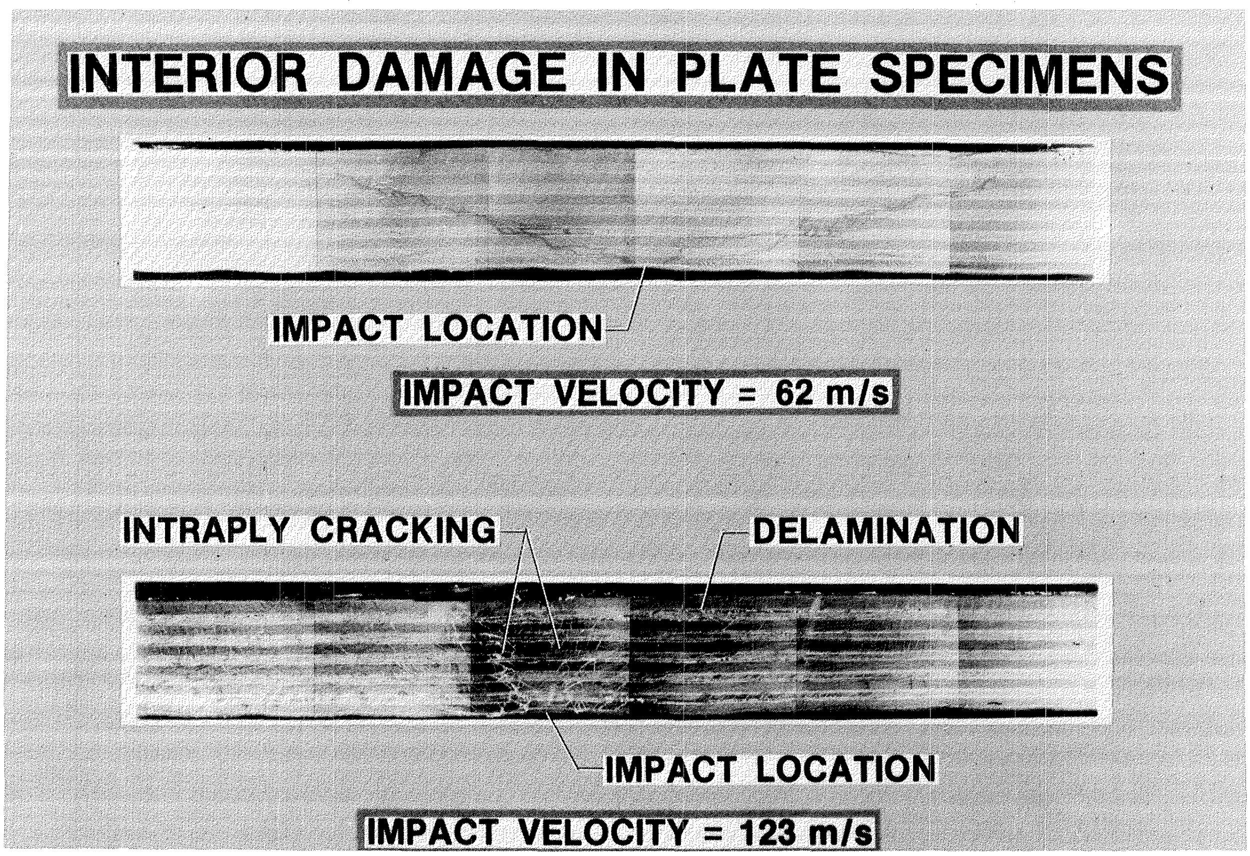

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Figure 7.- Cross sections of laminates damaged by impact (ref. 36).

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16. Abstract

As NASA enters the definition phase of the Space Station project, one of the important issues to be considered is structural material selection. The complexity of the Space Station and its long life requirement are two key factors which must be considered in the material selection process. Both aluminum and graphite/epoxy are considered as potential structural materials. This report presents advantages and disadvantages of these materials with respect to mechanical and thermal considerations, space environment, manufacturing, and cost.

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