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Honeycomb Sandwich Structures: Vented Versus Unvented Designs for Space Systems

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FOR THE COMMANDER

KURT A. JOHNSON, MAJOR, USAF
CHIEF, SYSTEMS ENGINEERING DIVISION

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Several space systems programs have experienced significant anomalies associated with failures of unvented honeycomb sandwich structures. Based on documentation and personal experience, this report provides descriptions of the structures involved, the circumstances of the failures, and corrective actions. Of special note, it is concluded that failures occur due to internal pressure when the external environment is reduced in ambient pressure. Such failures are initiated at weak bond areas (presumed to result from a manufacturing "defect" – areas that were not previously identified/repaired. Once initiated, separation of the skin-to-core bond propagates very rapidly and has been described as "explosive,"

It is recommended that honeycomb structures for space systems be adequately vented whenever possible. In situations where venting is not feasible, a series of recommendations is presented to minimize the likelihood of failure. Design and manufacturing parameters are listed for honeycomb sandwich panels that are not fully vented, in order to reduce the likelihood of failure.

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FOREWORD

The causes of failure or other anomalies generally can be described as:

- Incorrect design—service test conditions exceed design limitations
- Selection of the wrong material
- Inadequate material, process, or part specification
- Improper implementation of the specification
- Defective material—specification deficient
- Processing error—poor workmanship
- Inadequate quality control/inspection

The goal of every program is to minimize these problems so as to prevent failures/anomalies. One important way to accomplish this end is to be aware of previous problems and their resolutions/prevention strategies. This report is intended to provide such information as related to honeycomb sandwich structures as used in space systems.

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1. INTRODUCTION

Honeycomb sandwich structures are used in a wide variety of critical structures in Air Force space systems. These include payload fairings (shrouds) for launch vehicles, adapters for mounting of satellite payloads, solar array substrates, antennas, and equipment platforms.

Since 1964, there have been several known or suspected failures of honeycomb structures. These failures have been attributed to the lack of venting in the panel design/manufacture. On the other hand, based on available information, vented honeycomb sandwich panels never have experienced failure during flight. In the cases documented herein, the consequences of the failures have been significant and costly.

Honeycomb sandwich panels that are not vented will contain air (and possibly volatiles, including moisture) which causes a pressure differential during launch into orbit. If heating also is involved, the internal pressure will rise further. In any case, each individual unvented honeycomb cell acts as a tiny pressure vessel imposing stresses on the skin-to-core bonds. If these stresses are high enough, panel failure (i.e., skin-to-core debonding) will occur. Certain defects introduced during panel manufacture would make failure more likely.

This report is intended to document the known anomalies associated with unvented honeycomb sandwich structures. Because the consequences of such failures can be extremely severe (e.g., loss of mission and cost/schedule impacts), it is important to understand how such occurrences can be prevented.

Key factors affecting sensitivity to fabrication defects (i.e., the "robustness" of the design and manufacturing process) have not been determined. However, based on the documented information, it is possible to offer observations and concomitant recommendations. Further, it is advisable to identify the design/manufacturing parameters that are critical in cases in which unvented designs are considered necessary.

2. DEFINITIONS

A honeycomb sandwich structure is composed of a honeycomb core bonded to facings (skins).

Vented honeycomb sandwich structures use perforated, slotted, or porous honeycomb cores through which air can flow readily from cell to cell at a rate corresponding to the pressure drop during the ascent phase of a launch vehicle. Venting to the exterior is provided either through the skin or panel edge members. The changes in pressure within the panel ideally should occur at a rate corresponding to the external atmospheric pressure change during launch vehicle ascent.

An unvented honeycomb sandwich structure uses an unvented core (so that there is no air transfer from cell to cell) and/or does not provide for venting to the exterior of the panel. The use of a perforated honeycomb core by itself does not provide a vented structure; the panel must provide exterior venting as well.

Partially vented honeycomb structures provide some venting but not sufficient to accommodate the pressure differential, so some internal pressure is developed during ascent. Larger panels are more likely to be partially vented.

These conditions are illustrated in Figure 1.

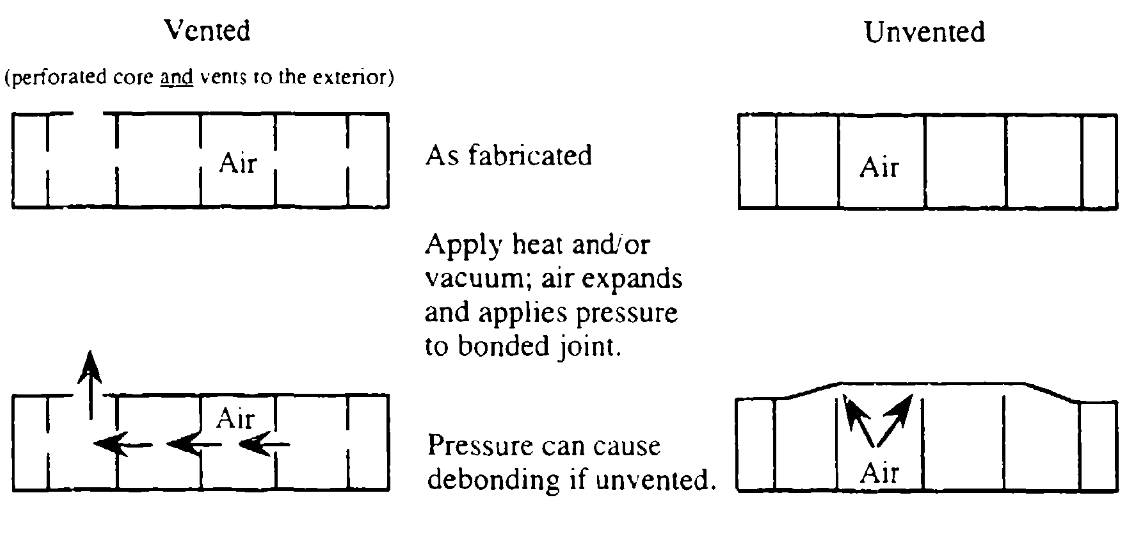


Figure 1. Vented versus Unvented Honeycomb Structures

3. HISTORICAL BACKGROUND

Table 1 describes the salient characteristics of honeycomb structures that are considered to have failed due to lack of venting. The following discussion is intended to review the circumstances related to each of these failures, including the findings from the various investigations that were reported.

3.1 MARINER 3

A fiberglass/phenolic honeycomb structure was used for the payload fairing for the National Aeronautics and Space Administration (Jet Propulsion Laboratory) [NASA(JPL)] Mariner 3, launched in 1964.

Anomalies observed in the acceleration rate gyro and fairing internal pressure data suggested that the Mariner 3 fairing failed at T + 103 sec (Ref. 1). Conditions at that time were: altitude, 95,000 ft; velocity, 4500 ft/sec; fairing maximum surface temperature, 600°F. Subsequent ground tests verified that the failure was attributed to "a pressure buildup in the fiberglass honeycomb core." (For Mariner 4, the payload fairing was redesigned using 0.090-in. thick magnesium-thorium alloy.)

Of special note, prior to flight a series of tests had been conducted to verify the original honeycomb design. These consisted of independent exposures to the predicted in-flight temperature and pressure environment—but not the combined pressure/temperature environment. After the flight failure, test panels were subjected to the combined environment. Failure occurred when an internal pressure of 26 psi and a surface temperature of 660°F were recorded. Another panel was tested with 0.03-in. diameter vent holes, drilled on 1.25-in. centers in the outer surface. This panel successfully withstood the combined temperature and altitude environment. However, a panel vented in this manner, but exposed to high humidity, failed at T + 106 sec when tested. (It is not known whether failure occurred because the venting was not sufficient to accommodate the additional pressure buildup due to the steam generated, or due to a defect in the panel.)

Table 1. Characteristics of Failed Honeycomb Structure

Program	Application	Design Considerations			Remarks
		Skins	Adhesive	Core	
Mariner 3	Payload fairing	0.020-in. glass/phenolic 0.040-in. glass/phenolic	HT-431 (epoxy/phenolic)	HRP, 3/16 in.* (glass/phenolic)	Both surfaces painted. Tmax (predicted) 675°F
Titan III	Payload fairing	0.023-in. glass/phenolic	HT-435 (epoxy/phenolic)	HRP, 3/8 in.* (glass/phenolic)	Surfaces unpainted. Tmax (predicted): 560-620°F
GPS	Solar array substrates	three plies, graphite/epoxy (Kapton film on one skin)	FM-24 (epoxy)	Aluminum, perforated	
Atlas-Centaur	Payload fairing	five plies, glass/phenolic (outer skin) four plies, glass/epoxy (inner skin)	HT-424 (epoxy/phenolic)	HRP, 3/8 in.* (glass/phenolic)	Sealed skins

*Cell size (diameter of inscribed circle)

A full-scale flight fairing was tested successfully under vacuum conditions *only* (no simulated aerodynamic heating). However, when it was exposed to the combined environment, failure occurred at T + 110 sec, at which time the surface temperature was 450°F. The test fairing failed suddenly and extensively in a manner that was described as "explosive." The apparent cause was separation (debonding) of the inner skin, resulting in inward collapse of the entire structure.

Flatwise tensile tests (2- x 2-in. specimens) were conducted (Ref. 1) to determine the skin-to-core bond strength. Values averaged 220 psi at room temperature, ranging from 180 to 252 psi (three test specimens). Two tests conducted at 400°F gave values of 87 and 94 psi. (The panels were cut from a production fairing which had a curvature radius of 37.5 in.; hence, the values measured are expected to be lower than those obtained from standard flat panels.) *Note that the stresses developed due to internal pressure sources are substantially lower than the skin-to-core (flatwise tensile) strengths, even at elevated temperature. From this, we can conclude that the panel failures probably were due to defects that escaped inspection, combined with the internal pressure.*

Peel tests (climbing drum, per ASTM D-1781) of samples cut from a payload fairing led to the conclusion that the peel strength "was reasonably close to that which would be expected for the honeycomb sandwich material used in the flight shrouds" (Ref. 1). Values ranged between 3.4 and 4.5 lb/in. width.

Permeability tests also were conducted of the honeycomb core. When subjected to vacuum conditions, pressure decay in the core was 9.6 torr/min parallel to the ribbon direction, and 2.1 torr/min perpendicular thereto. From this it was concluded (Ref. 1) that the rate at which air permeated the honeycomb (cell-to-cell) "was insignificant as compared with the ΔP which would be experienced during ascent through the atmosphere." *This means that there would be a pressure differential within the core, unless each cell were vented to the exterior through a skin. To effect a well-vented structure using vent holes along the panel edges would require vent holes (or slots) in one or more of the walls between the individual honeycomb cells.*

Based on the results of acoustic fatigue tests, JPL (Ref. 1) concluded that "acoustical excitation did not significantly contribute to the structural failure of the flight shroud."

Some of the conclusions from this extensive test program are particularly noteworthy:

a. The fiberglass/phenolic honeycomb cell-to-cell venting is insignificant compared to the pressure decay expected in flight.

b. Perforation (venting) of the skin would significantly reduce the pressure differential and would allow the shroud to survive flight.

c. Moisture within the honeycomb cells can add to the pressure differential, especially if heated.

3.2 TITAN III

In 1966, a group of satellites was to be launched by a Titan launch vehicle using a fiberglass/phenolic honeycomb sandwich payload fairing. During launch, the fairing exploded resulting in loss of all satellites. A remaining fairing of this type, already fabricated, was modified by drilling vent holes throughout the skin to permit venting. (Subsequently, a major program was undertaken to redesign the payload fairing so as to avoid the use of honeycomb sandwich construction.)

Because of concerns related to the Mariner 3 failure, tests were conducted to assess the Titan III payload fairing. A series of seven test panels was conditioned in a humidity chamber and then subjected to combined temperature and altitude environment. One panel failed, with the outer skin separating "explosively" from the core. At the time of the failure, T + 105 sec, conditions were: outer skin temperature, 700°F; altitude, 200,000 ft; and peak heat flux, 5.5 Btu/sec. (Internal pressure was not measured.)

Tests also were conducted of painted panels, preconditioned at high humidity, compared to "dry," unpainted panels. Maximum differential pressures were significantly different:

"Wet," Unpainted Panel -	0.5 - 1.0 psi
"Dry," Painted Panel -	12-16 psi

Based on these data, it was decided to "fly" with unpainted fairings. (Note that this permits the porous fiberglass/phenolic skins to serve as "vent holes," thereby reducing internal pressure.)

A thermal stress analysis for the fairing resulted in the following:

Assumed Skin Modulus (psi)	Maximum Outer Skin Compressive Stress (psi)
1.1×10^6	3790
3.1×10^6	5350
	Maximum Core Tensile Stress
	4.22

These results indicate that the stresses induced by aerodynamic heating are relatively low, not including stresses introduced by the internal pressure associated with the lack of venting. (*In other words, if there were adequate venting and no significant defects, the structure would not fail.*)

As part of the above investigation, it was recognized that the porosity of the laminate (glass/phenolic) skins can provide a degree of venting. It was stated that "porosity should fall within a specified range to assure that an adequate venting level is maintained;" however, no value/range was recommended. Further, because their "strength properties are lower than standard laminates, the use of a porous laminate for inner facing is not recommended."

The need for improving the process of skin-to-core bonding was noted, including the use of a "more reliable" nondestructive test (i.e., ultrasonic or Trepan testing), rather than the "tap" test. Also, a higher-density, smaller cell size (3/16 in. rather than 3/8 in.) honeycomb core was recommended to provide more bonding surface and higher sandwich strength properties. Processing to minimize water absorption during storage and on the launch pad also were considered important.

3.3 GLOBAL POSITIONING SYSTEM

For the Block II Global Positioning System (GPS) satellite. Rockwell International (RI), Seal Beach, California elected to design the solar array substrates using an unvented honeycomb design. (Block I used vented honeycomb panels.) The rationale was related to possible outgassing into the Space Shuttle cargo bay. Thin graphite/epoxy skins (three plies) were precured; one skin incorporated a thin Kapton film dielectric (to provide the surface to which the solar cells later would be attached). The skins were bonded to aluminum honeycomb core using a modified epoxy film adhesive.

In 1983, after RI finalized its stress and thermal analysis, there was a concern over the adequacy of the unvented design. Accordingly, to verify this design, RI conducted a thermal-vacuum development loads test on a panel. The panel ruptured during thermal cycling, and failure occurred at the skin-to-core bond. Interestingly, the failure occurred at the bond to the thinner skin (without Kapton film). As a result, RI decided to change to a vented honeycomb design and use a higher-temperature-resistant film adhesive, FM-300, in place of FM-24.

Inadvertently, a number of panels were made using nonperforated *unvented* honeycomb core for the outboard solar arrays. During the thermal-vacuum qualification test in October 1985, one such panel ruptured before completion of the first thermal cycle while it was being heated to 150°C. In December 1985, a second panel was tested and failed just after reaching 150°C. (*As a matter of interest, during attachment of solar cells, it was noted that these panels behaved differently than the usual solar array substrates. Apparently, deformation of the panel surface occurred due to internal pressure rise during processing operations.*)

Subsequently, these panels were scrapped, and the required solar array panels were remanufactured using vented honeycomb core and the FM-300 adhesive, with the panels vented along the edges.

3.4 ATLAS - CENTAUR PAYLOAD FAIRINGS

In April 1987, General Dynamics/Fort Worth conducted an "altitude" proof test of the AC-58 payload fairing intended as the "backup" for the AC-68 payload fairing that was to be used during launch of the FLTSATCOM F-8 spacecraft. (Note that, because of discrepancies previously observed, the AC-58 fairing had been dispositioned as "hold for Engineering Test.") The proof test had been introduced due to an anomaly that occurred several years earlier (1981) during launch of the FLTSATCOM F-5 spacecraft; at that time, it was postulated that the payload fairing (AC-59) may have exploded (disbonded) during ascent, causing extensive damage to the spacecraft. The "altitude" proof test consists of placing the structure (actually one-half of the fairing which is made in two halves) in a chamber and pumping down to simulate an altitude of 90,000 ft.

Figure 2 (from Ref. 2) shows the construction, consisting of a four-ply glass/epoxy inner skin and a five-ply glass/phenolic outer skin, bonded with HT-424 film (tape) adhesive to 1.75-in. thick HRP (glass/phenolic) honeycomb core of 3/8 in. cell size.

During the proof test of the AC-58 payload fairing, it exploded. High-speed motion pictures recorded the event. There was a small blisterlike separation observable in one frame, followed by rapid growth of an apparent disbond and explosive separation of the inner skin-to-core bond, accompanied by extensive fracture of the inner skin (see Figure 3; from Ref. 2).

Investigation indicated that the failure initiated in an area in which there had been a weak bond, i.e., there was no imprint from the core on the adhesive film. However, no discrepancies had been noted (i.e., no voids observed) in this particular area, based on numerous sonic ("tap") tests. (Many *other* areas had been found defective and were repaired by injecting a liquid epoxy resin to fill the discrepant/debonded areas.) Various possible causes were postulated for the "weak-bond" area:

a. Contamination of core or adhesive film surfaces

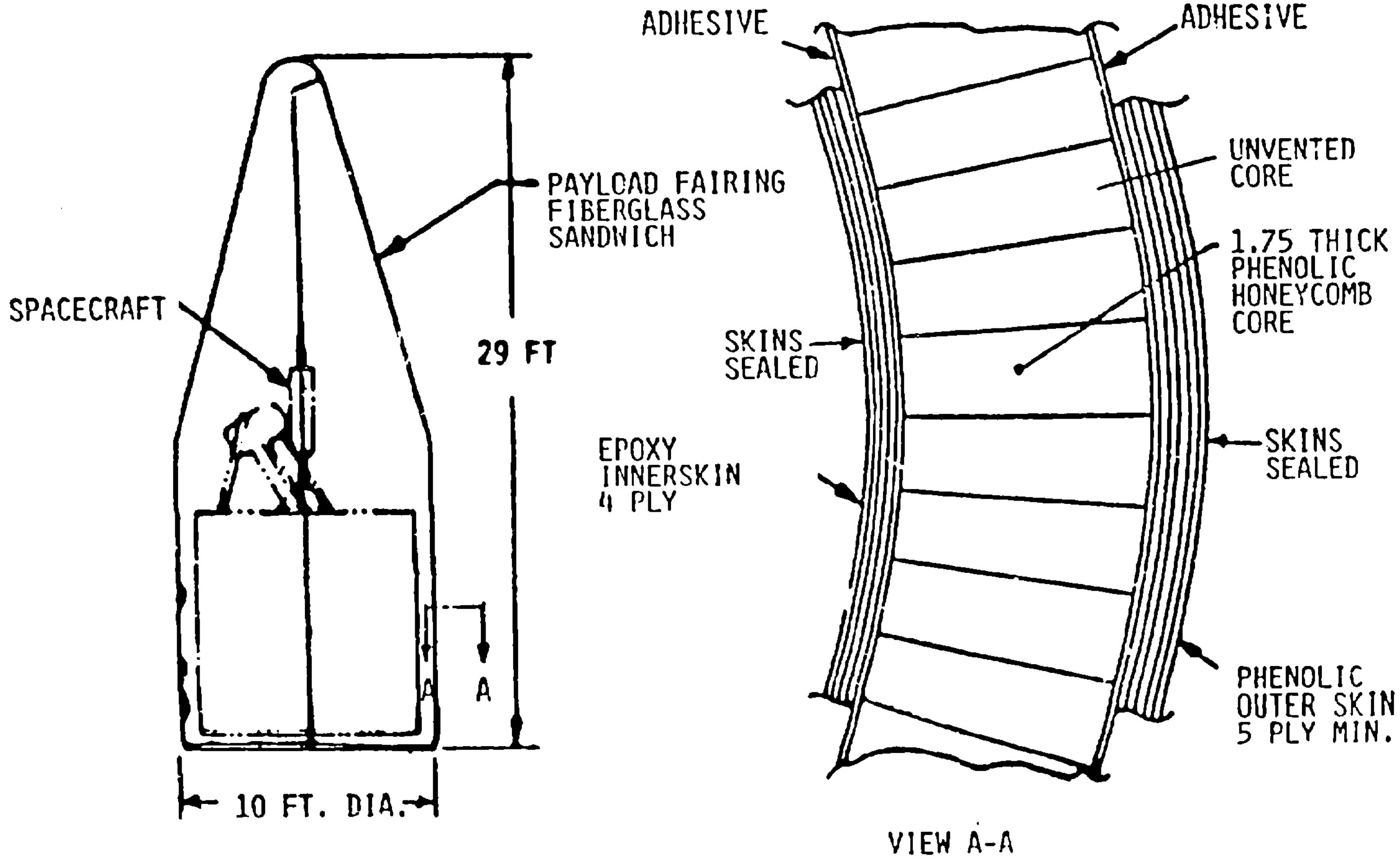
b. Inadequate flow of adhesive during cure

c. Cure pressure and/or temperature too low (affects adhesive flow and contact between core cell walls and skins)

d. Poor contact between mating surfaces

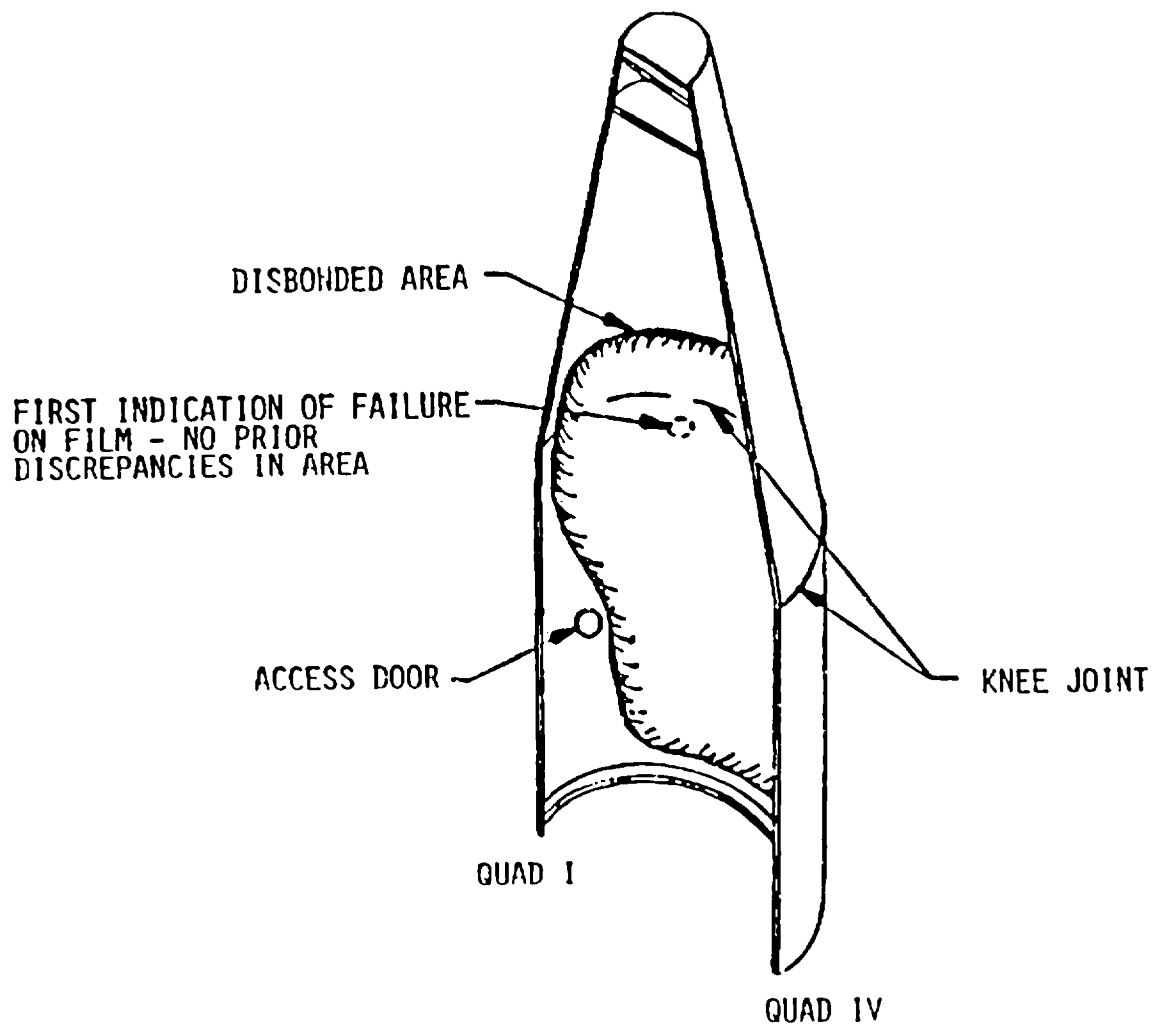
   1. Poor fit
   2. Cure pressure too low

Particularly noteworthy, the "tap" inspection had failed to detect the discrepancies in both AC-58 and AC-59 fairing failures; and both fairings had similar designs/processing, including nonvented honeycomb construction.



Courtesy of Engineering Directorate, NASA Lewis Research Center

Figure 2. Fiberglass Payload Fairing



Courtesy of Engineering Directorate, NASA Lewis Research Center

Figure 3. Spare Payload Fairing Uncapped Half Inner Skin Debond

In the recovery effort, it was decided to apply additional cork insulation on the exterior to reduce the glueline temperature and to conduct an overpressure proof test of the AC-68 payload fairing intended for flight. It was noted also that there had been improvements in the materials and processes for fairings produced after AC-59, i.e., for AC-60 and beyond. A thicker adhesive film was introduced, and more attention was paid to processing details and controls. Also the inner skin was increased from three to four plies.

The augmented overpressure, ambient-temperature, proof test was implemented successfully for the AC-68 payload fairing. The maximum expected pressure during flight was calculated to be 16.1 psi, allowing for pressure rise due to heating to 155°F at the bondline. Allowance also was made for bond strength reduction (8%) at elevated temperature (155°F), 1.3 psi; water partial pressure saturated at 155°F, 0.8 psi; and instrumentation error (5%), 0.9 psi. Accordingly, the overpressure proof test was conducted at ambient room temperature by pressurizing internally to 19.1 + 0.5/-0.0 psi using GN_2. (Subsequently the AC-68 payload fairing, with additional cork insulation on the exterior, was used successfully to help launch the FLTSATCOM F-8 satellite into orbit.)

For applications which involve elevated-temperature exposure, the reduction in bond strength is an important factor to be considered. For example, see Figure 4 which was used in establishing the overpressure allowances for the AC-68 payload fairing. Inasmuch as the overpressure proof test was conducted at ambient room temperature, the internal pressure applied was increased by a finite amount to accommodate the reduction in bond strength that would be expected at the maximum temperature during ascent.

In conducting the overpressure proof test, it was necessary to know the location of core splices which could impede GN_2 flow from cell to cell. The HRP core was porous to some extent, so that it was possible to pressurize selected areas of the payload fairing and achieve adequate uniformity. For the AC-68 payload fairing, a total of 53 pressurization ports and 14 sensing ports (pressure transducers) were employed. The test was conducted in an altitude chamber which was pumped down to 90,000 ft altitude (14 psid) to reduce the amount of GN_2 pressure required (and reduce the risk of damage to test equipment). The payload fairing was inspected nondestructively by X-ray and sonic techniques ("tap" test) before and after the pressurization test.

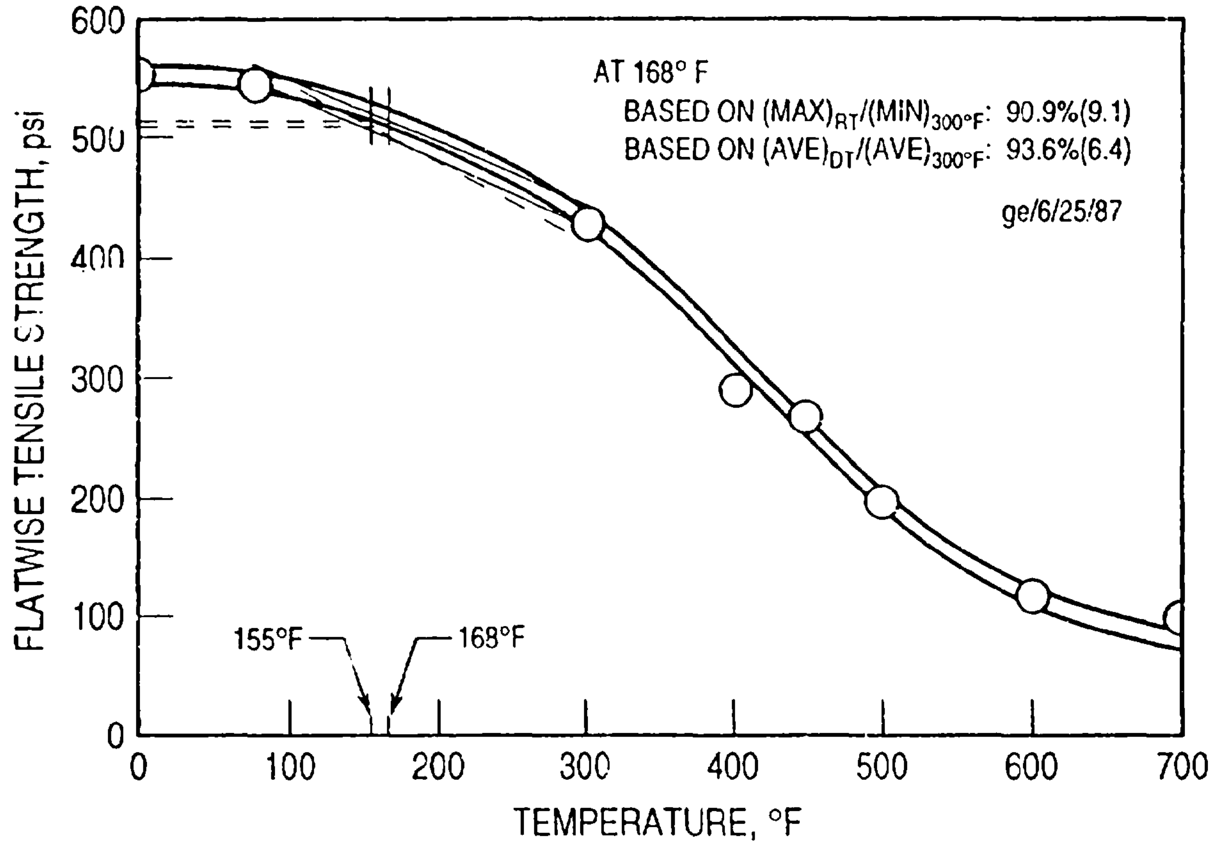


Figure 4. Typical Relationship Between Skin-to-Core Bond Strength and Test Temperatures

4. CONCLUSIONS AND RECOMMENDATIONS

Unless there is a special circumstance, honeycomb sandwich structures for space systems applications should be designed so that they are vented adequately (i.e., internal pressure essentially follows the exterior pressure environment so that there is practically no pressure differential). Exposure to high humidity should be avoided if possible or minimized.

In those cases in which venting is not feasible or may be less than optimum, the following recommendations should be followed:

a. The contractor should conduct an extensive series of development and verification tests.

b. Assuming such a program is successfully implemented, the fabrication of flight panels should be monitored by experienced Materials & Processing (M&P) personnel to help ensure that there are no production problems that escape identification.

c. Each part should be subjected to proof tests with nondestructive (NDT) inspections conducted before and after the tests. These should include exposure to reduced pressure and elevated temperatures, as applicable.

d. Special precautions should be taken to ensure that damage will not occur during operations such as handling, assembly, and so forth. Close surveillance would be required whenever the panels are not enclosed in well-protected containers.

Table 2 lists the parameters considered important to reduce the likelihood of failure in panels that are not fully vented. For the most part, these same considerations are appropriate in the design and manufacture of effective honeycomb sandwich structures.

In the event of a marginal or weak skin-to-core adhesive bond, the additional stress due to internal pressure over and above those normally encountered during launch/flight could serve as "the straw that breaks the camel's back." Especially in a large honeycomb sandwich structure, there always is the possibility of a localized area with a bond strength substantially below the average/norm. It is such areas of weak/marginal bond strength that are of concern and can lead to failure of the entire panel. For unvented honeycomb structures subject to internal pressurization, once a localized bond failure is initiated, further debonding propagates rapidly, leading to what has been described as "explosive" failure. Based on the current state of the art, it is not prudent to rely solely on nondestructive testing; proof tests are considered necessary (as discussed above), particularly if the honeycomb structure is not fully vented so as to ensure no pressure differential during ascent.

Table 2. Parameters for Reducing Likelihood of Failure in Not Fully Vented Honeycomb Panels

Design Parameters	Manufacturing Parameters
• Smaller cell size	• One-step cure/bonding for laminated skins
• Thicker/stiffer skins	• Prime core
• Thicker adhesive	• Verify fit up of skins and core
• High peel-strength adhesive	• Verify flow characteristics of adhesive at time of application/cure
• Reduce glueline temperature (external insulation)	• Apply maximum pressure during cure
• Fewer core splices	

Note: These same parameters, of course, also can be applied to vented honeycomb sandwich structures.

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