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APOLLO EXPERIENCE REPORT - SPACECRAFT STRUCTURAL WINDOWS

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16. Abstract The window structural design and verification experience is presented for the Apollo command and lunar modules. This report presents window design philosophy, design criteria, hardware description, and qualification and acceptance test programs and discusses the problems encountered and solutions developed in these areas. The structural characteristics of glass are not generally well understood by designers. The optics and instrument glass covers were not considered to be structural components and thus were not normally subjected to the design, qualification, and acceptance standards necessary to preclude failures. These two factors contributed significantly to window problems on both Apollo spacecraft.			
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SUMMARY

The Apollo command module and lunar module window structural design and verification experience is presented. The design philosophy, design criteria, hardware, qualification and acceptance tests, problems, and problem resolutions are discussed.

Instrument windows and guidance and navigation optics were not considered to be structural items during their design and had to be structurally verified late in the program. In addition to the lack of a good window definition, which contributed to inadequate structural design and verification rigor, a general misunderstanding of the structural characteristics of glass by design engineers was noted.

A structural window definition is proposed to identify glass structures that should be verified flightworthy. A spacecraft window is structurally defined as any piece of glass that is thermally or mechanically stressed and will endanger either the crew or mission success if it breaks.

Fracture mechanics is the best means known today to understand, explain, and evaluate the structural characteristics of glass and to provide a structural rationale for specifying proof-test requirements. Use of fracture mechanics in all future window designs and evaluations is proposed to ensure acceptable window designs and to eliminate the misunderstanding of the structural characteristics of glass.

INTRODUCTION

The significant aspects of the structural design and verification of the Apollo command module (CM) and lunar module (LM) windows are presented in this report. Design philosophy and criteria, hardware configuration, qualification testing, acceptance testing, and problems encountered (with resolutions) are discussed. Also presented are brief discussions of the structural characteristics of glass and of fracture mechanics analysis of glass. Information concerning Apollo window contamination is given in reference 1.

A spacecraft window is structurally defined as any piece of glass that is thermally or mechanically stressed and will endanger the crew or mission success if it breaks. This definition was not used during the design of the CM and LM. The lack of definition contributed to inadequate structural design and verification of some glass items that are not normally considered structural.

Nine CM windows and four LM windows were integral parts of the Apollo spacecraft primary pressure vessels (habitable volumes). Also, there were many glass instrument covers in the CM and LM that were not part of the primary pressure vessels but which sealed and protected the instruments from the spacecraft environment. The structural integrity of these windows affected crew safety and mission success to varying degrees. Windows and glass structures were not treated as a separate technology during the Apollo spacecraft design and development; therefore, consistent design philosophy and design criteria were not used initially throughout the program. This situation was recognized late in the Apollo Program during a NASA Lyndon B. Johnson Space Center (JSC), formerly the Manned Spacecraft Center (MSC), review of glass structural characteristics and the analytical tools for design, analysis, and verification of glass structure. It was also determined that the structural characteristics of glass were not well understood by window designers. This review identified fracture mechanics as the best analytical means available to evaluate the structural integrity of glass and to provide a structural rationale for specifying proof-test requirements. Fracture mechanics techniques were used to reevaluate the glass structure in the CM and LM; this method resulted in some modifications and some requalification and reverification tests.

STRUCTURAL CHARACTERISTICS OF GLASS

Both tempered and annealed glass were used for Apollo spacecraft windows. A brief discussion of the structural characteristics of glass is included to introduce the CM and LM structural design philosophy and structural design criteria.

Essentially, glass is a noncrystalline, supercooled liquid that is isotropic and elastic. Two conditions, tensile stress and surface flaws, are required for a fracture in glass, which has an inherent strength (without flaws) in the range of 1 to 3 million psi. All bulk glass, such as window material, has microflaws caused by manufacture and handling that reduce its strength. In addition, visually undetectable flaws through the thickness can be present in annealed glass, which necessitates a rigorous flaw-screening program.

Annealed glass exhibits a characteristic called static fatigue, which is a degradation of allowable stress as a function of time at load. Static fatigue is flaw growth caused by a combination of stress and environment and is sometimes referred to as stress corrosion. Water or moisture is a prime contributor to stress corrosion of glass.

Tempering in glass is a process that puts the surface in compression, thus eliminating surface tensile stresses. Thermal tempering results in a parabolic residual stress distribution through the glass thickness with a compressive stress on the external surface approximately twice that of the tensile stress at the middle of the glass.

Chemical tempering has a much higher compression-to-tension ratio, but the compressive layer is much thinner than with thermal tempering.

The strength of both tempered and annealed glass is defined by the modulus of rupture (MOR), which is the short-time breaking stress of glass caused by bending in a moist environment. The MOR is generated for each type of glass by using abraded test samples to decrease the data scatter. The MOR values are used for design by the glass industry for both annealed and tempered glass. The type of abrasion is assumed to represent the worst flaw to be expected in the glass. The MOR of tempered glass consists of the residual stress in the glass at the flaw tip plus the inherent strength of the glass if it were annealed. The lower limit strength of polished samples is approximately the same as for abraded samples.

COMMAND MODULE WINDOWS

The CM has five double-pane windows: one hatch window, two side windows, and two rendezvous windows (fig. 1). Each of these windows consists of inner-pressure-vessel and heat-shield windows as shown in figures 2, 3, and 4. Four single-pane windows were located in the CM guidance and navigation (G&N) optics. In this report, the CM windows are separated into four categories: inner structure, heat shield, G&N, and instrument windows. Because the G&N and instrument window problems were similar for the CM and LM, they are discussed in a section that applies to both.

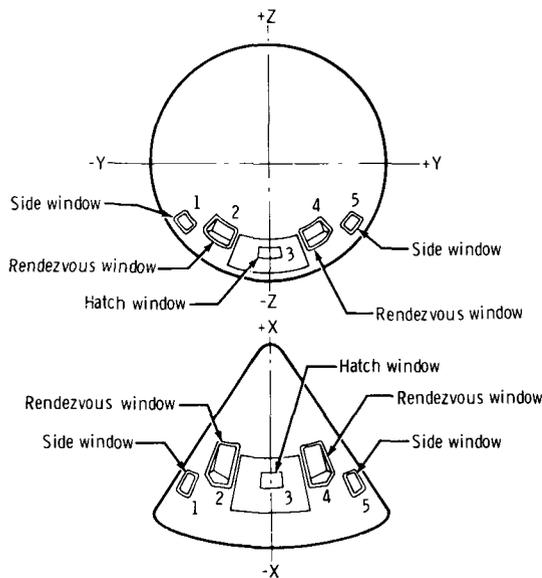


Figure 1. - Command module window designation.

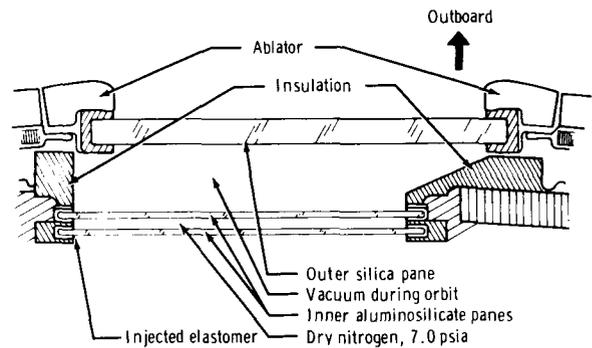


Figure 2. - Command module side windows (1 and 5).

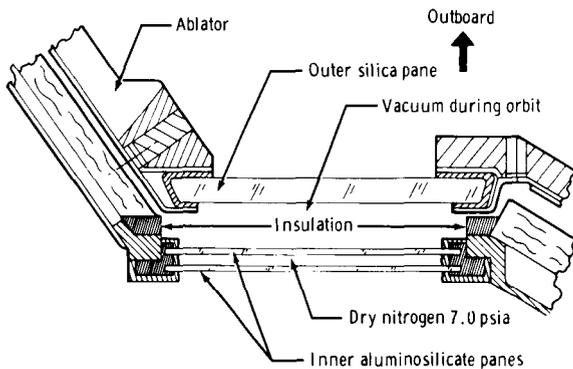


Figure 3. - Command module rendezvous windows (2 and 4).

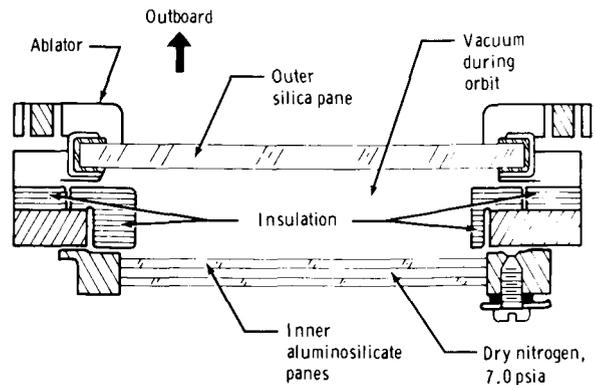


Figure 4. - Command module hatch window (3).

Inner-Structure Windows

Design philosophy. - The structural design philosophy adopted for the CM inner structure, which is the CM primary pressure vessel, was that all windows would be double-pane windows for redundancy and made of tempered glass. The cavity between the two panes would be evacuated and backfilled to 7.0 psia with a dry inert gas. The windows were to be mounted to preclude installation stresses, thus restricting the glass loading to differential pressure. The frames were to be designed to restrict the window loads to those induced by pressure on the window.

Design criteria. - The structural design criteria used for the Apollo CM inner-structure windows were different from those used throughout the aerospace industry, including those used for the Mercury and Gemini spacecraft. In general, a factor of safety of 3.0 was required, based on the MOR of the glass used. Also, the limit pressure was usually the maximum pressure that could exist during the mission, excluding any failures. The factor of safety required for the CM inner-structure windows was 1.5, based on the residual compressive stress in the glass caused by tempering. The CM windows were thus designed so that the glass surfaces had zero tension stress at ultimate load. Because the CM windows were double pane and the cavity between the panes was at less than atmospheric pressure, the possibility existed that the windows could have developed a prelaunch leak and be subjected to 14.7 psid limit pressure in space, assuming a check-valve leak. Because the pressure in the window cavity was not verified after installation, the criterion was adopted that the design limit pressure would take into account a check-valve seal leakage failure. Therefore, all CM double-pane windows were designed for a 14.7 psid limit pressure across each pane.

Description. - All CM inner-structure windows were made of aluminosilicate glass and were thermally tempered to 25 000 psi MOR for the hatch and side windows and to 23 200 psi MOR for the rendezvous window. The two panes in each of the windows had the same thickness: 0.23, 0.25, and 0.20 inch for the hatch, side, and rendezvous windows, respectively. To accomplish the stress-free mounting and to provide a seal, a silicone elastomer was injected around the edge of each pane and cured in place. This construction essentially potted the windows in their frames. After the seal had been cured, the volume between the double-pane inner-structure window was evacuated

and backfilled with dry nitrogen to 7.0 psia. Each inner-structure window was coated on both sides with a high-efficiency, antireflection (HEA) coating (fig. 5).

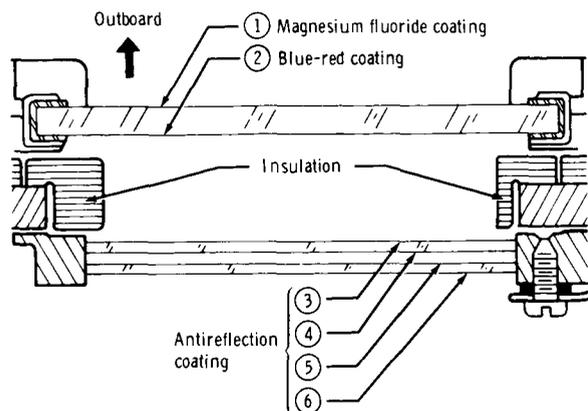


Figure 5. - Command module window-surface notation.

Qualification. - The qualification tests for the inner-structure windows consisted primarily of strength and deflection tests and mission-life pressure-leak tests. The strength tests were conducted on windows abraded to simulate the maximum expected surface flaws. The design limit pressure for the inner-structure windows was 14.7 psid, which assumed a check-valve leak of one of the window seals. Five strength tests were performed on each configuration, the last of which was a failure test. The pressures at failure for the hatch window, side window, and rendezvous window were 29.5, 29.7, and 32.5 psid, respectively. These strength tests were performed on abraded, uncoated windows. It was assumed that the coatings did not affect the window strength. This assumption

later was proved invalid, and additional qualification tests to determine the effects of the coatings were required. These tests are discussed in the section of this report entitled "Problems."

Heat-Shield Windows

Design philosophy. - The initial structural design philosophy for the CM heat-shield windows was that all windows subjected to the entry environment would be single-pane windows made of annealed, fused silica glass. However, an outer pane, also of annealed fused silica, was provided to protect each heat-shield window from micrometeoroid impact while in space. These micrometeoroid windows were not required to withstand the entry environment. Later in the program, the micrometeoroid windows were omitted to reduce weight.

Design criteria. - The primary structural design criterion for the CM heat-shield windows was a factor of safety of 1.5, based on the sandblasted, annealed, fused silica MOR degraded for the time at load. The heat-shield windows are exposed to stress for approximately 15 minutes during entry, and the primary loading is thermal. The time used to determine the allowable strength (considering flaw growth) was 1 hour. The criterion generally required by the aerospace industry is a design factor of safety of 3 with a proof factor of 2, based on the MOR of abraded (sandblasted) glass. The structural degradation of glass caused by stress and humidity can cause problems during proof testing that can be misinterpreted. Also, the proof test can contribute to flaw growth. The structural criterion for annealed windows in the CM was a lower factor of safety based on a more realistic structural allowable and a limit stress that included worst-case conditions.

Description. - Each of the five CM heat-shield windows consisted of a 0.7-inch-thick single pane made of fused amorphous silica. Each pane was insulated around the edge by a thermal insulator encapsulated by a 0.02-inch-thick Fiberglas layer (using a silicone elastomer bonding agent). The insulation was flexible enough to allow the edges of the glass to rotate. The 0.080-inch-thick 17-4 PH steel frame and retainer were designed so that the flat glazing could be installed on the conical heat shield. A 0.05-inch gap was provided on all edges between the insulation and frame to allow the shell to contract in the cold-soak environment without loading the glass. Each heat-shield window was coated as shown in figure 5.

Qualification. - The qualification tests for the heat-shield windows consisted of strength tests for the appropriate abort design conditions and a thermal test of the design entry condition from an initial cold-soak condition. Limit pressure tests to -5.5 and +11.14 psid and ultimate pressure tests to -8.3 and +16.7 psid were conducted on each of the three window configurations. The analytical and test-temperature differentials between the inner and outer surfaces of the heat-shield window were 1600°, 930°, and 1130° F for the hatch, rendezvous, and side windows, respectively.

Quality Control

To screen flaws, each of the Apollo CM windows was subjected to an acceptance program. The acceptance tests for each window included thermal-shock tests, pressure tests, and visual inspections.

The thermal-shock test of the heat-shield windows consisted of heating the windows in an oven to 1200° F and quenching in 68° to 77° F water. The thermal-shock test of the inner-structure windows consisted of heating the windows in a salt bath to 500° F and quenching in 68° to 77° F water. Although the stresses in a window caused by the thermal-shock test are difficult to predict, the test screens flaws over the entire surface of the window, which is important for heat-shield windows because of the thermal loading.

Each inner-structure and heat-shield window pane was subjected to a proof-pressure acceptance test to screen flaws. The proof-test pressure for all inner-structure windows was 22 psi. For the heat-shield windows, the pressures were 49 psi for the hatch window, 40 psi for the side window, and 73 psi for the rendezvous window. Windows were visually inspected to ensure that they were not damaged subsequent to the proof test. It was assumed that the surface would have to be visibly damaged for the window to be significantly flawed.

The CM window proof-test requirements were reevaluated subsequent to the MSC study that determined a general lack of understanding of glass structural characteristics among window designers. The proof-test requirements were reevaluated using fracture mechanics technology (ref. 2) and rational analysis techniques. Although proof tests for some conditions were not of the magnitude that would be used if the windows were being designed (specifically, a proof test to 1.5 of the operating stress instead of a value derived from fracture mechanics), the windows were determined to be acceptable for crew safety and mission success.

Problems

As previously stated, an initial assumption was made that the optical coatings applied to the windows did not degrade their structural capability; therefore, the window qualification tests were conducted using uncoated, abraded windows. This assumption was proved invalid during coating crazing anomaly, and additional qualification testing of the coated windows was required to verify their flightworthiness.

To improve the optical properties of the CM heat-shield hatch window for the lunar multispectral photography experiment (S-158) on Apollo missions 13, 14, and 15, the available hatch windows were surveyed to find those with acceptable optical properties. Two heat-shield side windows with acceptable properties were found, and it was decided to remove the coatings and cut hatch windows from these side windows. At that time, the hatch scientific window could have no coatings because of the S-158 experiment optical requirements. The "no coating" requirement was removed later for the Apollo 14 mission because the S-158 experiment on that flight was rescheduled to the Apollo 15 mission. The first side window was damaged during the polishing procedure to remove the coatings. The second window was cut to size, and both the magnesium fluoride and the blue-red (BR) coatings were left on the window. In the thermal-shock test, the window was heated in air to 1200° F for 45 minutes and immediately immersed in room-temperature tapwater. The BR coating crazed and cracks propagated through the tension layer developed by the thermal gradient of the glass on the underside of the BR coating. Later, an entire side window from spacecraft 014 was tested in a similar manner to verify that the cutting of the window had not caused the crazing; similar results were obtained.

In August 1970, three coupon tests were conducted, one of which was thermally shocked in room temperature water and two of which were slow cooled in air. The following conclusions were reached.

1. The BR coating crazing is temperature dependent.
2. The BR coating crazing is independent of window size or cooling rate.
3. The BR coating crazing occurs during cooling.
4. Window glass fracture sometimes occurs when the BR coating crazes, but does not occur without it.

To determine the minimum temperature at which the heat-shield window coating crazes, a series of tests was conducted on 2- by 7-inch samples cut from a spacecraft 017 heat-shield side window. The results of these tests indicated that the threshold temperature for BR coating crazing is 450° to 500° F. A separate series of tests was conducted on 3- by 3-inch samples cut from a spacecraft 014 heat-shield rendezvous window and also determined that BR coating crazing occurs at approximately 500° F.

The decision was made to continue using the coated windows and to conduct a delta qualification test program to verify the adequacy of these windows at newly predicted flight temperatures. The maximum flight temperature on the inner surface of the heat-shield window was predicted to be 362° F. Because the coating crazing was

determined to be caused by peak temperature and not thermal gradient, it was proposed that the heat-shield windows be tested to a maximum temperature of 462° F to account for variations in oven temperature and the temperature at which the coating was deposited. Three heat-shield side windows were heated in an oven to 462° F to represent worst-case flight temperatures, and the windows were then pressure tested on both sides to verify adequate structural capability for the remaining flight loads. The program was completed successfully and the coated heat-shield windows were considered flight qualified.

A test program was also conducted to verify that the coating process did not significantly degrade the structural capability of the tempered glass windows. Three hatch windows were pressurized to 23 psig on each side separately to verify that the coating process did not degrade the windows and that the windows would pass the acceptance test. The three hatch windows were pressure tested to failure, which occurred at 47, 42, and 41 psig in a humid environment.

LUNAR MODULE WINDOWS

The LM has four windows in its primary structure. The two forward windows are located in the cabin front-face bulkhead. The docking window is located in the upper section of the cabin wall. Each window contains two panes with the cavity between the panes vented to the external environment. In this configuration, only the inner pane of each window was subjected to the cabin pressure loading. The fourth window in the primary structure is the G&N window (optical telescope) and is discussed in another section.

Design Philosophy

The structural design philosophy for the LM windows was to provide a window of minimum weight with maximum crew visibility, which led to the selection of the single-pane-window concept using chemically tempered glass. The design consisted of a single structural pane and an external pane for micrometeoroid and radiation protection.

Design Criteria

The basic structural design requirement was that the windows had to sustain the environment imposed on them by the cabin with the primary loads attributable to the pressure and thermal environment. The design limit pressure, the maximum cabin relief valve pressure, was 5.8 psid. The design ultimate factor of safety was 2.0. However, the minimum acceptable MOR bending modulus (strength) of the glass, subsequent to all manufacturing processes, was specified to be 50 000 psi. This strength corresponds to a factor of safety of approximately 7.2, based on the stress at design-limit pressure. The thermal environment imposed on the windows was +350° to -90° F.

Description

The ascent stage was configured with three windows as shown in figure 6. The two 25- by 28- by 24-inch triangular forward windows located in the front-face bulk-head of the forward cabin section were installed in the vehicle in a plane oblique to the vehicle axes to provide an approximate visibility of 65° down and 80° outboard. These windows provided the required visibility during the lunar descent, lunar landing, and lunar stay phases of the mission. Each window consists of two panes separated by a cavity vented to the external space environment. A cross section of the edge of this window is shown in figure 7. The outer (nonstructural) pane was a micrometeoroid- and radiation-protective window made of annealed glass. This material was chosen for its excellent optical quality, low coefficient of thermal expansion, and high resistance to hypervelocity impacts. The inner pane was the structural window made from chemically tempered glass. This material was chosen for its high strength-to-weight ratio, high resistance to low-velocity impacts, low coefficient of thermal expansion, good optical quality, and good thermal-shock characteristics. The outer pane was clamped to the window frame, whereas the inner pane was a "floating" (simply supported) pane mounted on a seal constructed from a metallic spring surrounded by a Teflon jacket.

An overhead docking window is located on the left side of the vehicle directly over the commander's head. The window provides the commander with the required visibility during the initial phase of the descent to the lunar surface and during the final phase of the docking maneuver. The docking window has approximately 65 square inches (5 by 13 inches) of viewing area. The construction of this rectangular window, shown in figure 8, was similar to that of the forward windows. One exception was that its inner structural pane was not floating; it was bolted to the cabin skin by a metallic edge member bonded to the chemically tempered glass. An additional difference was that the inner pane of the docking window was curved to match the 92-inch diameter of the cabin instead of being flat as the forward

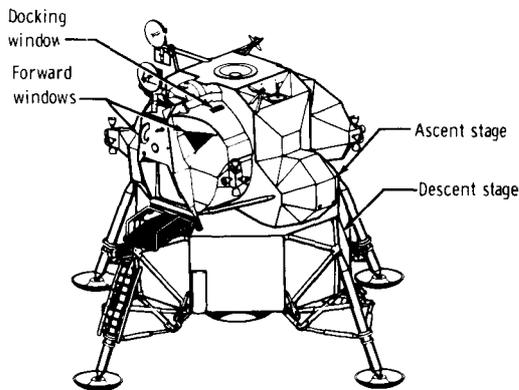


Figure 6. - Lunar module window location.

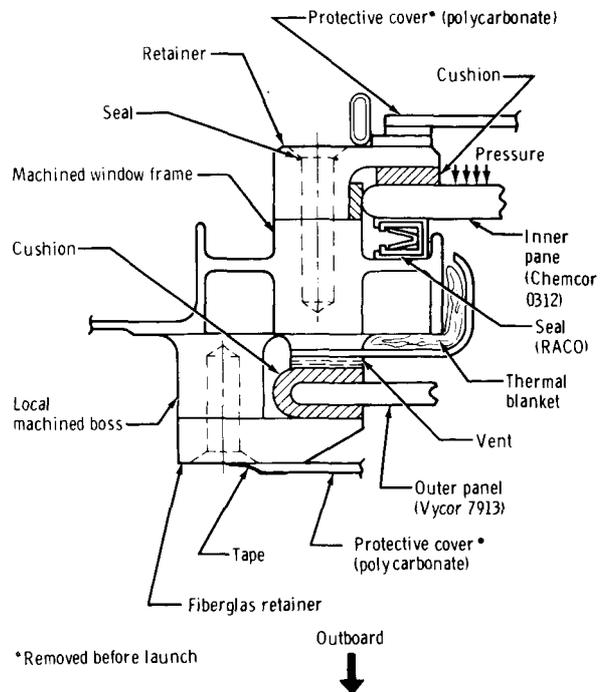
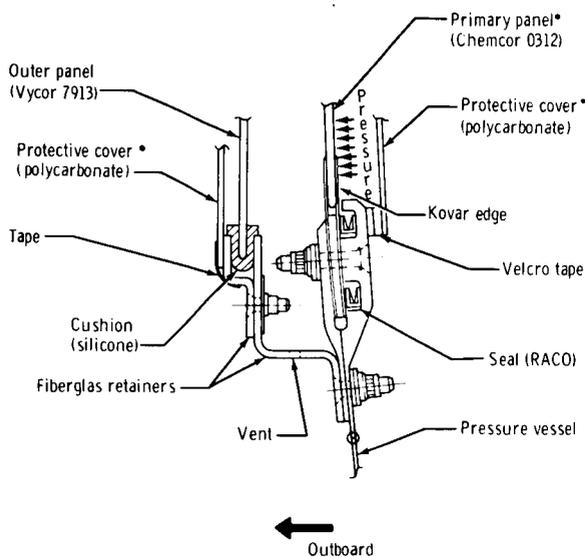


Figure 7. - Cross section of edge of LM forward window.

windows are. The forward windows are floating and carry only pressure loads. The forward windows were designed to support the cabin pressure as a plate. The docking window is integral to the cabin structure and is therefore subjected to all loads supported by the cabin in addition to the pressure loads. The docking window was designed to support the cabin loads as a membrane. However, the cabin loads in the vicinity of the window (because of other applied loads) are small when compared to the pressure load.

The LM windows had three different coatings applied to the glass (fig. 9). An electrical conductive coating (ECC) was applied to the outboard surface of each inner pane. An electrical connection was made to the silver bus bar on each side of the window to provide the ECC current. The bus bars on the forward windows were powered at 45 to 76 watts to defog the panel. The inner docking window was smaller and therefore required only 18 to 24 watts to defog the panel. The ECC was applied evenly on the docking window but unevenly on the forward window to obtain the required electrical power and thermal dissipation needed to defog the panels. The approximate thickness of the ECC was 400 to 700 angstroms for the forward window and 2500 angstroms for the docking window. The original light transmission of the chemically tempered glass before the ECC was applied was approximately 88 percent. After ECC application, light transmission was reduced to approximately 76 percent. To increase the light transmission and decrease the reflection caused by the ECC, an HEA coating was applied to the inboard surface of each inner and outer pane. When a pane was completely coated and a black edge (black velvet paint) was applied to the periphery, the light transmission was increased to approximately 82 percent and the reflection was reduced from approximately 14 percent to 5 percent. A final BR coating was applied to the outer surface of each outer pane to restrict the amount of infrared and ultraviolet light to the cabin.



* Removed before launch

Figure 8. - Cross section of LM upper docking window.

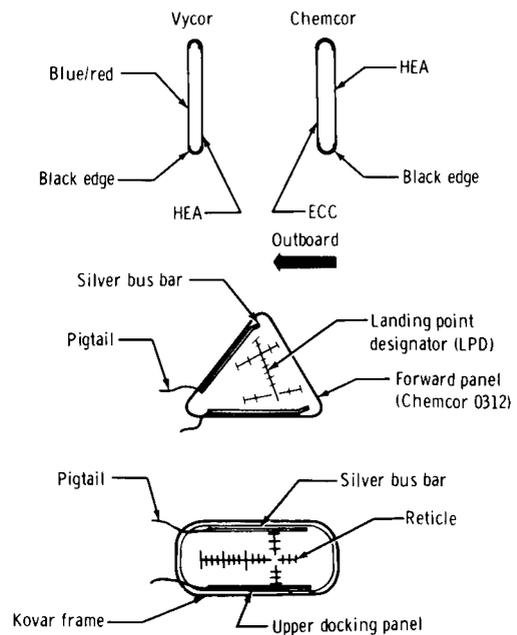


Figure 9. - Lunar module window coatings.

Qualification

The LM windows were originally qualified for space flight at both the component and vehicle level. The component level tests consisted of both functional and structural tests. Functional tests showed the optical and electrical aspects of the windows, whereas the structural tests included tests for leakage, shock, acoustics, thermal-vacuum, and humidity.

The vehicle level tests were accomplished as part of the LM structural vehicle test program. In this test program, the windows were exposed to design environment conditions, including pressure, vibration, and landing loads. No problems concerning the windows were encountered during the component or vehicle level tests.

Problems

Forward window test failure. - In December 1967 during a factory checkout pressure test on the LM-5 ascent stage cabin, a forward window failed at a cabin pressure of 5.1 psid. The pane was completely destroyed, which is consistent with the failure mode of tempered glass. The subsequent investigation revealed no abnormal conditions; it was concluded that the most likely cause of the failure was an undetected flaw in the glass. This pressure test of LM-5 was the first time the window glass had been exposed to the correct stress distribution. Previous structural acceptance tests on the windows consisted of a mechanical flexure test (fig. 10) on the forward windows and a thermal-shock test on the docking window. The mechanical flexure test was accomplished three times on both sides of each panel (each corner on roller 1). The flexure test stressed the pane to a maximum of approximately 30 000 psi, which demonstrated a factor of safety of approximately 5 over normal operating stress. However, this test did not

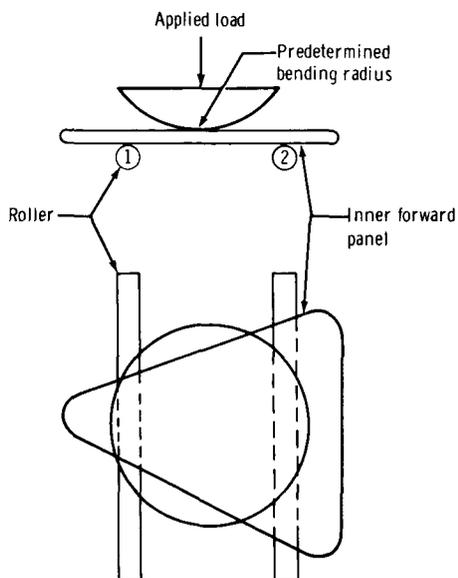


Figure 10. - Lunar module window flexure tests.

stress the corners of the pane, which were the areas of maximum stress in the vehicle. Because the docking window was curved, a thermal-shock acceptance test was required instead of a mechanical flexure test. The stress level experienced during thermal shock was approximately 20 000 psi, which demonstrated a factor of safety of approximately 7. To reestablish confidence in the structural integrity of the LM windows, many engineering evaluation tests were conducted. A summary of the tests conducted, including the purpose, condition, and results, is given in table I. These tests showed the structural integrity of the window glass and the window-support structure. The structural integrity was sufficient if glass with critical defects was not used in the spacecraft. Two steps were taken to guarantee that defective glass was not installed. The first step was to modify the acceptance test to include a pressure test on each structural pane in a test fixture simulating the

TABLE I. - LUNAR MODULE WINDOW TEST RESULTS

Test	Purpose	Condition	Results
Stress data	To establish stress pattern on forward inner panel.	Pressurized with gaseous nitrogen from 0 to 25 psid.	Panel underwent stress of approximately $\pm 36\ 000$ psi at the three corners when pressurized to 25 psid.
Ultimate pressure test	To determine the pressure required to fail an inner forward panel. Three vehicle and five rejected panels were used.	Pressurized with water until failure.	Panel failure occurred at a range from 78 to 96 psid.
Fatigue test on docking panel	To determine the structural integrity of the inner docking panel when subjected to a series of pressure cycles.	Ten cycles to 15 psid. 500 cycles to 5.8 psid. Ten cycles to 15 psid. Taken to destruction. Pressurized with water. Cycle starts at 0 psid.	Panel survived cycling test. During the destruct test, panel failed at 55 psid.
Fatigue test on forward panel	To determine the structural integrity of the forward inner panel when subjected to a series of pressure cycles.	Ten cycles to 25 psid. 500 cycles to 5.8 psid. Ten cycles to 25 psid. Taken to destruction. Pressurized with water. Cycle starts at 0 psid.	Panel survived cycling test. During the destruct test, panel failed at 71 psid.
Impact test on forward panel	To determine the energy level required to fail an inner forward panel when subjected to a dynamic impact.	Vacuum fixture to produce a 5.8 psid. Impact panel at various speeds and locations. Weight - 11.6 lb with a protruding 1/8 in. radius point.	Panel survived impact at lower velocity. Failure occurred at the corner on the 18th impact with a velocity of 9 ft/sec (14.8 ft/lb).
Impact test on docking panel	To determine the energy level required to fail an inner docking panel when subjected to a dynamic impact.	Vacuum fixture to produce a 5.8 psid. Impact panel at various speeds and locations. Weight - 1.63 lb with a protruding 1/8 in. radius point.	Panel survived impact at lower velocity. Failure occurred on the 13th impact with a velocity of 8 ft/sec (1.63 ft/lb).
Scratch test	To determine the relation of glass strength to scratch size.	Specimen was scratched at 1/2-lb increments using a 3-mil radius point traveling at 12.5 linear inches per minute. Specimens were then taken to destruction. At 5 lb, the diamond point chipped. Specimen size - 0.190 by 1 by 10 in.	Failure occurred at edge on the 1/2- to 3-1/2-lb scratch. No decrease in MOR value (average 82 830 psi). Failure occurred at scratch location on the 4- and 4-1/2-lb scratch specimen. Decrease in MOR value (average 70 200 psi, 4-lb specimens and 25 800, 4-1/2-lb specimens).
Thermal shock on docking panel	To demonstrate the structural integrity of the inner docking panel after it has been subjected to a rapid change in temperature.	Panel was heated to 350° F for 20 minutes, then dropped into a 65° F water bath.	Panel survived test. Panel destruct at 64 psid.

TABLE I. - LUNAR MODULE WINDOW TEST RESULTS - Concluded

Test	Purpose	Condition	Results
Thermal shock on forward panel	To compare the structural integrity between chemically tempered glass and tempered soda-lime glass.	Panels were heated in a kiln temperature of 464° F for a period of 20 minutes. Panels were dropped into a 64° F water bath then pressurized to failure.	Chemically tempered glass survived test. Soda-lime glass developed spalls and cracks on edge. Chemically tempered glass panel failed at 22 psid. On another run, results were identical. Chemically tempered glass failed at 85 psid.
Temperature cycling on forward and docking panels	To demonstrate the ability of the inner panels to sustain a thermal cycling at a constant 7.7 psid.	Panels were subjected to thermal cycling, hot and cold soaks, for 220 hours between -90° F and 350° F.	Panels survived test, which simulated all phases of the LM flight including lunar stay.
Oxygen compatibility	To demonstrate the structural integrity of a forward window when subjected to combined environment of oxygen, elevated temperatures, high humidity, and pressure.	The panels were exposed to a temperature of 225° ± 5° F and a 90 to 100 percent relative humidity in a pure oxygen atmosphere. The test was conducted with a differential pressure of 25.0 psid for 72 hours and 11.6 psid for 144 hours.	Panels survived tests. Panels destructed at 90 psid and 82 psid.

spacecraft installation. The acceptance test pressure imposed on the panes was 25 psid for the forward windows and 15 psid for the docking window. These pressures correspond to proof factors of 5 and 3 above operating pressure for the two windows, and these proof factors are consistent with requirements based on fracture mechanics techniques. The docking window test pressure was restricted to 15 psid to prevent yielding of the metallic edge member. In addition to the improved acceptance tests, the second step taken was to redesign the window glass protective covers to provide maximum protection from acceptance until launch. Before the acceptance tests were conducted, a polycarbonate protective cover, retainer, and glass pane were installed as a subassembly in the pressure-test fixture. The pressure test was then performed to the proper pressure for 2 to 5 minutes. After the pressure test, the subassembly was removed from the fixture and an additional protective cover was installed on the opposite side to provide a handling and transportation protective cover assembly. The outer covers remained on each pane until the outer pane was installed. The two covers were not removed from the vehicle until just before launch, which provided maximum protection against inadvertent damage from workmen in and around the vehicle.

No vehicle windows have failed since the revised acceptance test procedures and new protective covers were implemented. Eight windows (seven forward and one docking) have failed in acceptance tests. The range of failing pressure on the forward window was 5.0 to 24.8 psid. The only docking window failure occurred at 11.6 psid.

Window heater electrical connection. - A forward window bus bar arced during the factory checkout test of the electrical heater on LM-8. Failure analysis revealed the glass was spalled under the bus bar, causing the arc when electrical power was applied. A detailed inspection conducted on other LM windows revealed discolorations that indicated spalling.

The configuration of the electrical connection to the bus bar for the window heater is shown in figure 11. The bus bar was a strip of silver paste fused into the glass along the edge of two sides of each window. The electrical wire was subsequently soldered to the bus bar. To provide strain relief to the solder connection, an overlay of epoxy was applied. After the addition of the solder and epoxy to the windows, the window was exposed to various manufacturing processes that elevated the window temperature. Because of the difference in the thermal coefficient of expansion between the glass, solder, and epoxy, residual stresses were induced into the glass. Analysis indicated that the magnitude of thermal stress was sufficient to relieve the compressive stresses in the outer layer of the chemically tempered glass, thereby reducing the strength of the glass. An extensive test program was conducted to verify the cause of the spalling. The MOR test matrix is presented in table II.

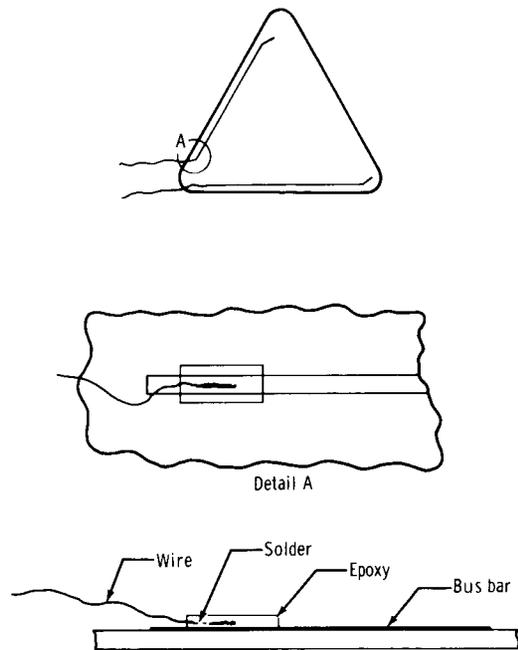


Figure 11. - Lunar module window heater electrical connection.

TABLE II. - LUNAR MODULE GLASS MOR TEST SPECIMEN MATRIX

	Group							
	I	II	III	IV	V	VI	VII	VIII
ECC and bus bar	X	X	X	X	X	X	X	X
Solder wire to bus bar		X	X	X	X	X		X
9-lb pull test on wire		X		X		X		X
Add epoxy to soldered joint		X	X			X		X
Hot soak at 250° F, 1 hr		X	X	X	X		X	X
Hot soak at 150° F, 1 hr						X		
Flex test three times at 18 367 psi		X	X	X	X	X	X	X
Cold soak at -10° F, 1 hr		X	X	X	X	X	X	X
Remove epoxy and wire								X
MOR average destruct test, psi	88 490	48 210	40 150	39 240	74 230	53 130	85 710	87 020

The test program, conducted on 35 MOR test bars, consisted of control samples (without solder and epoxy) and various combinations of solder, epoxy, and electrical wire pull tests. The samples were exposed to various thermal and flexure environments and then loaded to failure. The test results (table III) indicate a significant reduction in strength for the bars with the solder and epoxy attached and substantiate the stress analysis. Because the problem was caused by the presence of epoxy and solder, it was decided to redesign the electrical connection to eliminate any potential source of strength degradation caused by induced residual stresses or flaw generators. The test

TABLE III. - RESULTS OF TEST PROGRAM FOR LM FORWARD WINDOW HEATER BUS BAR FAILURE

[Proof pressure of 25 psid equivalent to 32 000 psi stress.]

MOR bar	ECC	Bus bar	Solder	Pull test	Epoxy overcoat	Temperature extremes, ° F	Flexure	Failure stress, psi	Remarks
1	X	X				Room temperature		85 000	Control
2	X	X				Room temperature		89 000	Control
3	X	X				Room temperature		92 000	Control
4	X	X				Room temperature		85 000	Control
5	X	X	X	X	X	250 to -10	X	76 000	
6	X	X	X	X	X	250 to -10	X	28 000	
7	X	X	X	X	X	250 to -10	X	66 000	
8	X	X	X	X	X	250 to -10	X	21 000	
9	X	X	X		X	250 to -10	X	41 000	
10	X	X	X		X	250 to -10	X	43 000	
11	X	X	X		X	250 to -10	X	46 000	
12	X	X	X		X	250 to -10	X	31 000	
13	X	X	X	X		250 to -10	X	38 000	
14	X	X	X	X		250 to -10	X	49 000	
15	X	X	X	X		250 to -10	X	37 000	
16	X	X	X	X		250 to -10	X	39 000	
17	X	X	X	X		250 to -10	X	30 000	
18	X	X	X			250 to -10	X	18 000	
19	X	X	X			250 to -10	X	70 000	
20	X	X	X			250 to -10	X	78 000	
21	X	X	X	X	X	150 to -10	X	31 000	
22	X	X	X	X	X	150 to -10	X	59 000	
23	X	X	X	X	X	150 to -10	X	48 000	
24	X	X	X	X	X	150 to -10	X	73 000	
25	X	X				250 to -10	X	83 000	Control
26	X	X				250 to -10	X	86 000	Control
27	X	X				250 to -10	X	87 000	Control
28	X	X	X	X	X	250 to -10	X	89 000	Strip epoxy, unsolder lead, and etch spalls on specimens 28 to 35
29	X	X	X	X	X	250 to -10	X	89 000	
30	X	X	X	X	X	250 to -10	X	93 000	
31	X	X	X	X	X	250 to -10	X	85 000	
32	X	X	X	X	X	250 to -10	X	86 000	
33	X	X	X	X	X	250 to -10	X	81 000	
34	X	X	X	X	X	250 to -10	X	82 000	
35	X	X	X	X	X	250 to -10	X	85 000	

program included specimens with solder and epoxy applied that were exposed to the flexure and temperature environment and then the solder and epoxy were removed. These specimens showed that the original strength was reestablished if the solder and epoxy were removed.

The electrical connection to the bus bar was redesigned to eliminate the solder and epoxy. The new electrical connection was achieved by a beryllium copper spring-loaded electrical contact as shown in figures 12 and 13. The window panes in LM-7 and subsequent vehicles were removed from the vehicles and the solder and epoxy were removed from the panes. Following the rework, the panes were proof-pressure tested and reinstalled in the vehicles.

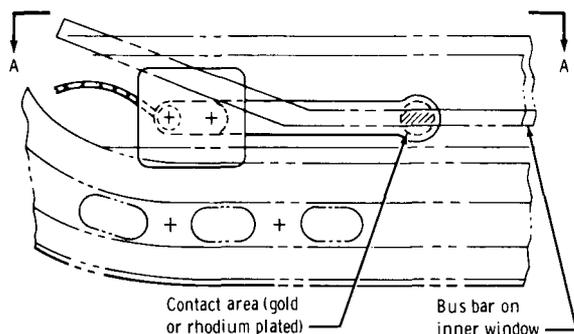
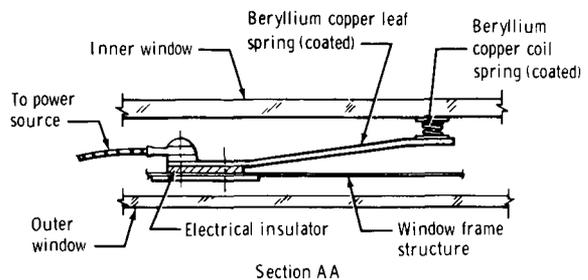


Figure 12.- Revised forward window heater electrical connection.

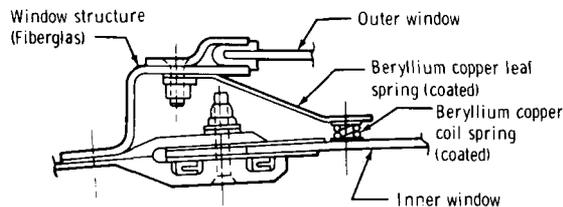
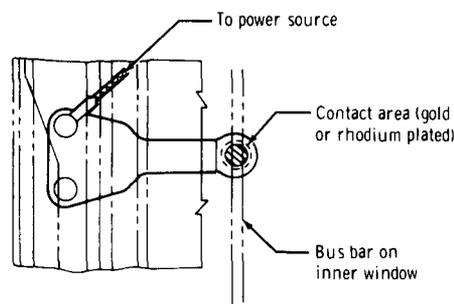


Figure 13.- Revised docking window heater electrical connection.

GUIDANCE AND NAVIGATION WINDOWS

There were four single-pane windows in the CM optical unit assembly and one single-pane window in the LM alignment optical telescope. These windows were not considered part of the structural subsystem during the structural qualification of the CM and LM because they were located in the optics of the G&N system. These windows did not receive rigorous structural qualification and acceptance testing consistent with the other windows because they were defined as optics rather than windows. In general, there was a lack or decrease in structural qualification and acceptance test rigor when glass was used in design by disciplines other than structures. For future programs,

all structural glass used in spacecraft windows or instruments should be evaluated and verified structurally by cognizant personnel.

The optical unit and the alinement telescope windows were reviewed, and a fracture mechanics analysis of each window was performed. The windows were determined to be structurally acceptable based on the leak test (14.7 psid) that had been performed. The threshold flaw size was greater than the thickness for each window, which indicated that these G&N windows would not fail catastrophically at the maximum operating pressure of the CM and LM. It was also demonstrated that the leak rate of the CM would remain within the emergency limits of the environmental control system if one of the pressure windows in the G&N optics system did fail catastrophically.

INSTRUMENT WINDOWS

During the initial inflight inspection of the LM on the Apollo 15 mission, the range/range-rate instrument window was found to be broken. This instrument and all other instruments on the LM and CM are hermetically sealed with 1 atmosphere (14.7 psia) internal pressure. Most of the LM and CM instrument windows, including the range indicator, are made of annealed soda-lime glass. The range indicator glass operating stress (sustained load caused by differential pressure) was in excess of the allowable for static fatigue and therefore was the probable cause of the failure.

Further investigation revealed additional instrument windows in both the LM and CM that were under excessive sustained stress and that had been exposed to inadequate proof test to ensure no failure by flaw growth. The approach used to remedy the inadequate CM and LM instrument window designs was to install permanent covers or doublers over the highest stressed instrument windows and temporary covers over some of the lower stressed windows to protect the crew if a window failed. The probability of no window failure for each instrument window was calculated based on MOR test data from the National Bureau of Standards. The effect of a failure of each instrument window from flammability and operational standpoints was also evaluated.

FRACTURE MECHANICS ANALYSIS OF GLASS

A comprehensive introduction to fracture mechanics analysis is presented in reference 2. Although not mentioned in reference 2, fracture mechanics is the best means known today to understand and evaluate the structural characteristics of glass. The fracture mechanics techniques for analyzing glass are similar to those for analyzing metals presented in reference 2.

It is not generally recognized among design engineers that the strength of glass is a function of environment and flaw size. Also not generally recognized is the fact that flaws not visually detectable can be critical. Fracture mechanics provides a rational basis for specifying the proof test required to verify that no flaws are present that could induce failure at operating stresses during a mission.

Some of the limitations of the fracture mechanics analysis are discussed in reference 3. Because fracture mechanics is not generally recognized as a method for analysis of glass, limited glass fracture mechanics data are available. Wiederhorn at the National Bureau of Standards is one of the few experimenters who has published glass fracture mechanics data (ref. 4). Fracture mechanics data can vary with glass composition and environment and should be measured for the specific composition and environment being evaluated.

The fracture mechanics analytical techniques and proof-test methods presently developed are provided in reference 2. Some material data and confirmation that fracture mechanics is a valid technique for analysis of glass are provided in references 2 and 3. All proof tests of glass should be performed in a dry or vacuum environment to reduce or eliminate flaw growth during the proof test.

CONCLUSIONS

The experience gained and the problems encountered in the design and verification of Apollo spacecraft windows demonstrate that there is a general lack of understanding of glass structural characteristics by design engineers and that there is a need for adequate structural criteria. The following items should be included in future window structural requirements.

1. All spacecraft windows and glass structures, including optics and instrument windows, should be structurally verified.
2. The rationale for proving the structural integrity of glass should include a proof acceptance test to screen flaws. Fracture mechanics should be used to analyze glass that must carry stress, and acceptance proof tests should be performed based on the fracture mechanics analysis.
3. Coatings and bonded attachments tend to degrade the structural integrity of glass and their use should be avoided. However, when coatings and bonded attachments are required, the windows should be thoroughly qualified in flight configuration and environments.

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National Aeronautics and Space Administration
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914-13-20-13-72

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