FEASIBILITY ASSESSMENT OF THERMAL BARRIERS FOR RSRM NOZZLE JOINT LOCATIONS

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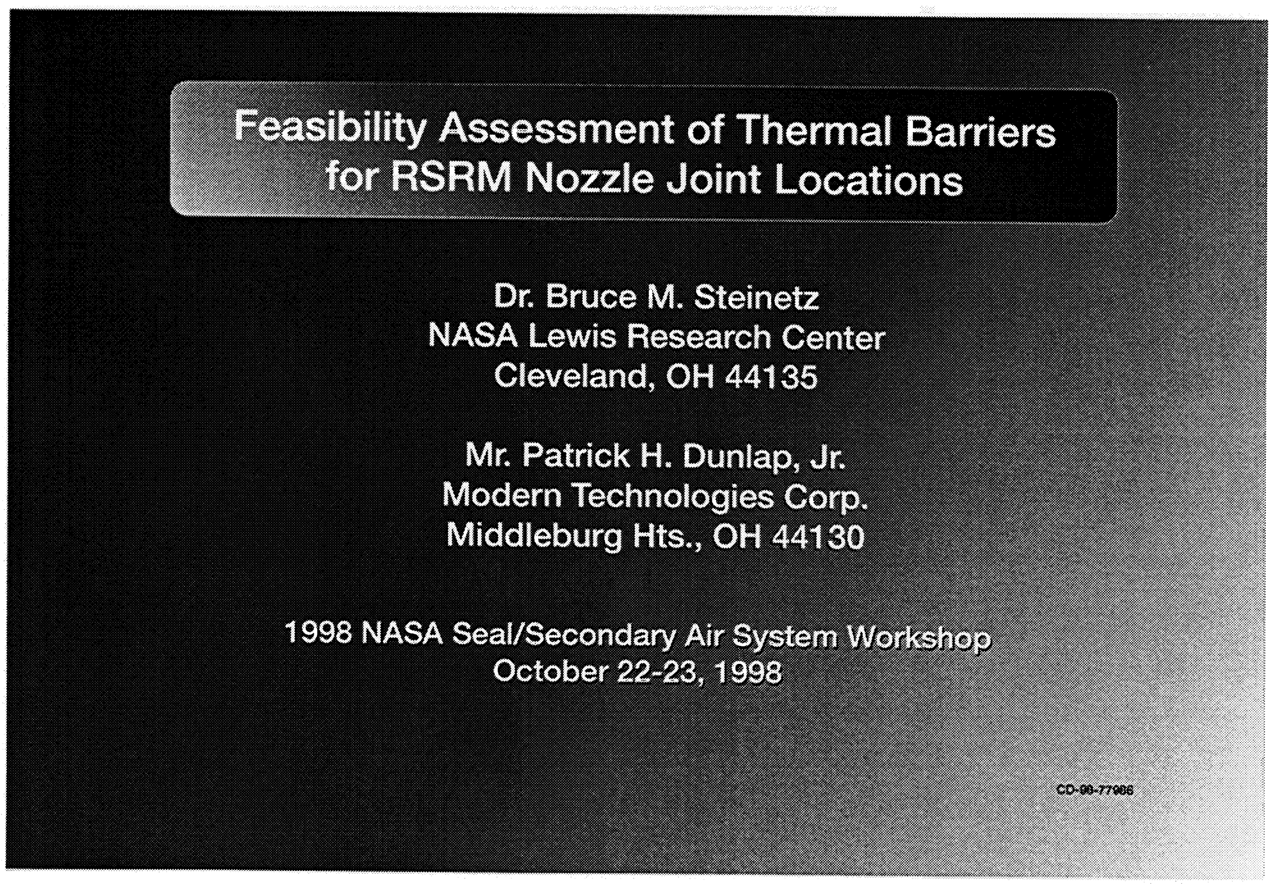
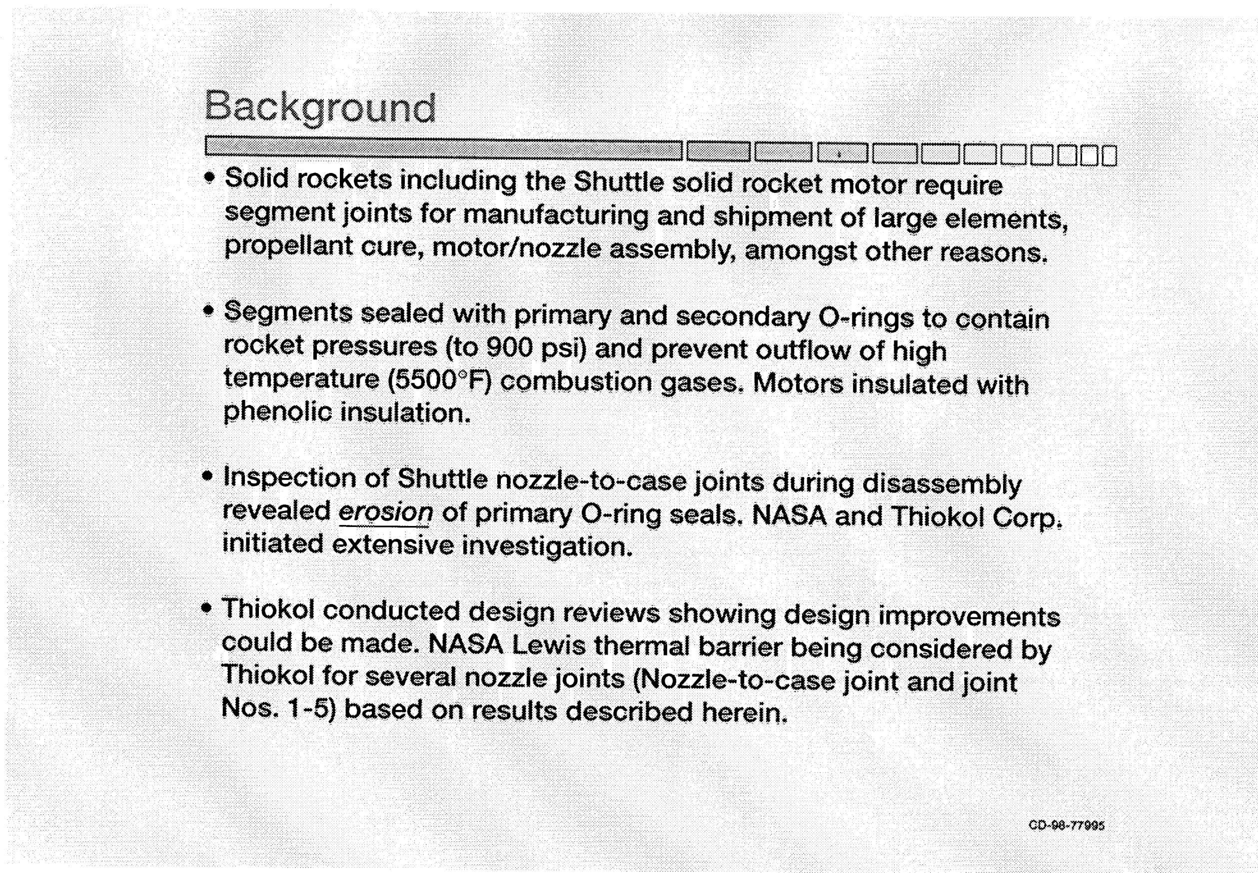
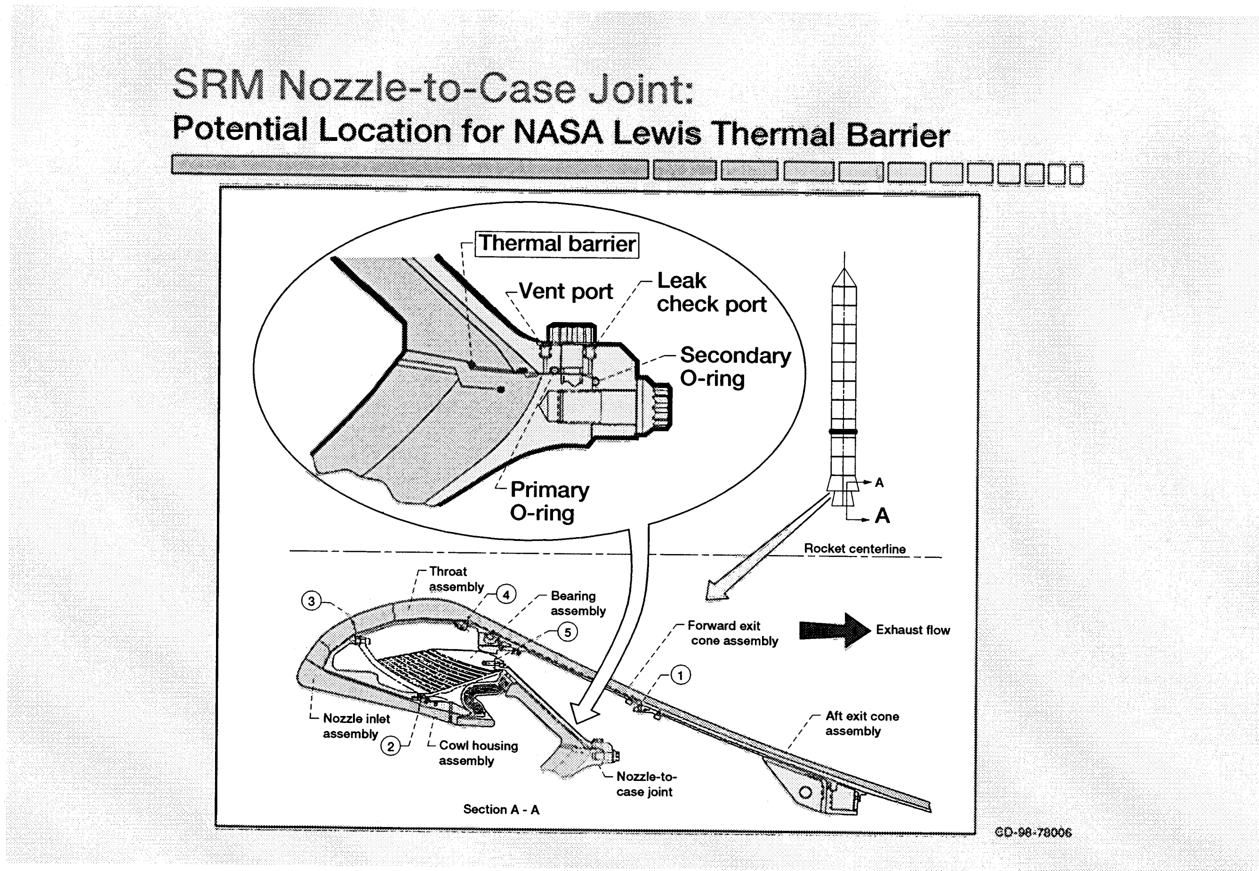




Solid rockets, including the Space Shuttle solid rocket motor, are generally manufactured in large segments which are then shipped to their final destination where they are assembled. These large segments are sealed with a system of primary and secondary O-rings to contain combustion gases inside the rocket which are at pressures of up to 900 psi and temperatures of up to 5500°F. The seals are protected from hot combustion gases by thick layers of phenolic insulation and by joint-filling compounds between these layers. Recently, though, routine inspections of nozzle-to-case joints in the Shuttle solid rocket motors during disassembly revealed erosion of the primary O-rings. Jets of hot gas leaked through gaps in the joint-filling compound between the layers of insulation and impinged on the O-rings. This is not supposed to take place, so NASA and Thiokol, the manufacturer of the rockets, initiated an investigation and found that design improvements could be made in this joint. One such improvement would involve using NASA Lewis braided thermal barriers as another level of protection for the O-ring seals against the hot combustion gases.



This chart shows where the thermal barrier would be used in the nozzle-to-case joint of the Shuttle solid rocket motor. The figure at the bottom is an enlarged area of the rocket nozzle showing the nozzle-to-case joint as well as nozzle joints one through five. The thermal barrier is also being considered for use in several of these other nozzle joints. The figure at the top is an enlarged view of the nozzle-to-case joint. The primary and secondary O-rings are shown along with the phenolic insulation (in orange) and the surrounding metal hardware (in blue). The thermal barrier is highlighted in its position upstream of the O-rings where it would help block hot combustion gases from reaching the O-rings.

Thermal Barrier Has Unique Requirements

- Sustain extreme temperatures (2500-5500°F) during Shuttle solid rocket motor burn (2 min. 4 sec.) without loss of integrity.
- Block hot flow gases from impinging on primary/secondary O-rings to prevent O-ring char or erosion.
- Exhibit some permeability to allow pressure check of primary/secondary O-ring system without any "false-positives" of primary O-ring seal.
- Exhibit adequate resiliency/springback to accommodate limited joint movement/separation and manufacturing tolerances in these large nozzle segments (diam. range 4.8' to 8.5').
- Endure storage for ≥ 2 years

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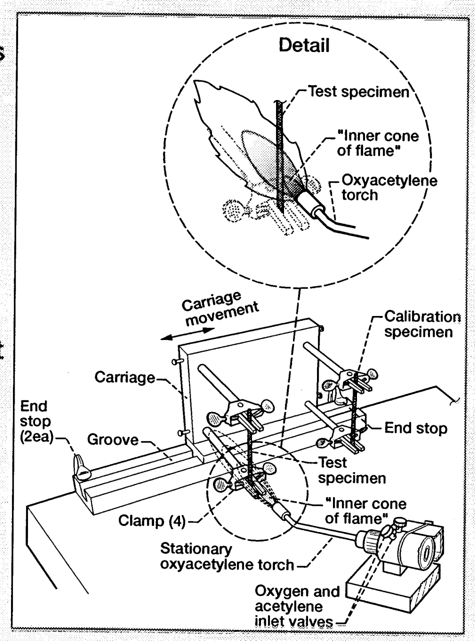
To be used in the nozzle-to-case joint of the Shuttle solid rocket motor, the thermal barrier has several unique requirements. It must be able to withstand extreme temperatures of up to 5500°F during the Shuttle solid rocket motor burn time of 2 minutes 4 seconds without loss of integrity. It must be able to block hot flow gases from impinging on the primary/secondary O-ring system to prevent the O-rings from becoming charred or eroded. However, the thermal barrier still has to be permeable enough to allow a pressure check of the O-rings without any "false-positives" of the primary O-ring. (Refer back to Chart 3) As shown in the schematic of the nozzle-to-case joint, the O-rings are pressurized using a leak check port between them to make sure that they are not damaged and are providing a good seal. If the thermal barrier provided a perfect seal and did not allow some gas to flow through it, it is possible that the primary O-ring could be damaged, but the thermal barrier would seal the joint and make it appear that the primary O-ring was sealing properly. Thus, some gas flow through the thermal barrier is required to prevent this from happening. (Return to Chart 4) The thermal barrier also must exhibit adequate resiliency to accommodate limited joint movement and separation and to make up for manufacturing tolerances in these large nozzle segments which have diameters of 4.8 to 8.5 feet. Finally, the thermal barrier has to be able to endure storage for as long as two years. Once the rockets are assembled, they often sit for several years before they are used.

Burn Test Apparatus and Procedure

- Candidate thermal barrier materials subjected to oxyacetylene torch "neutral" flame (5500°F) temperatures.
- Flame parameters "pre-set" using calibration specimen.
- Carriage fixture ensures consistent location of specimen in hottest part of flame for repeat tests.
- Time for "burn-through" measured for each candidate thermal barrier material.



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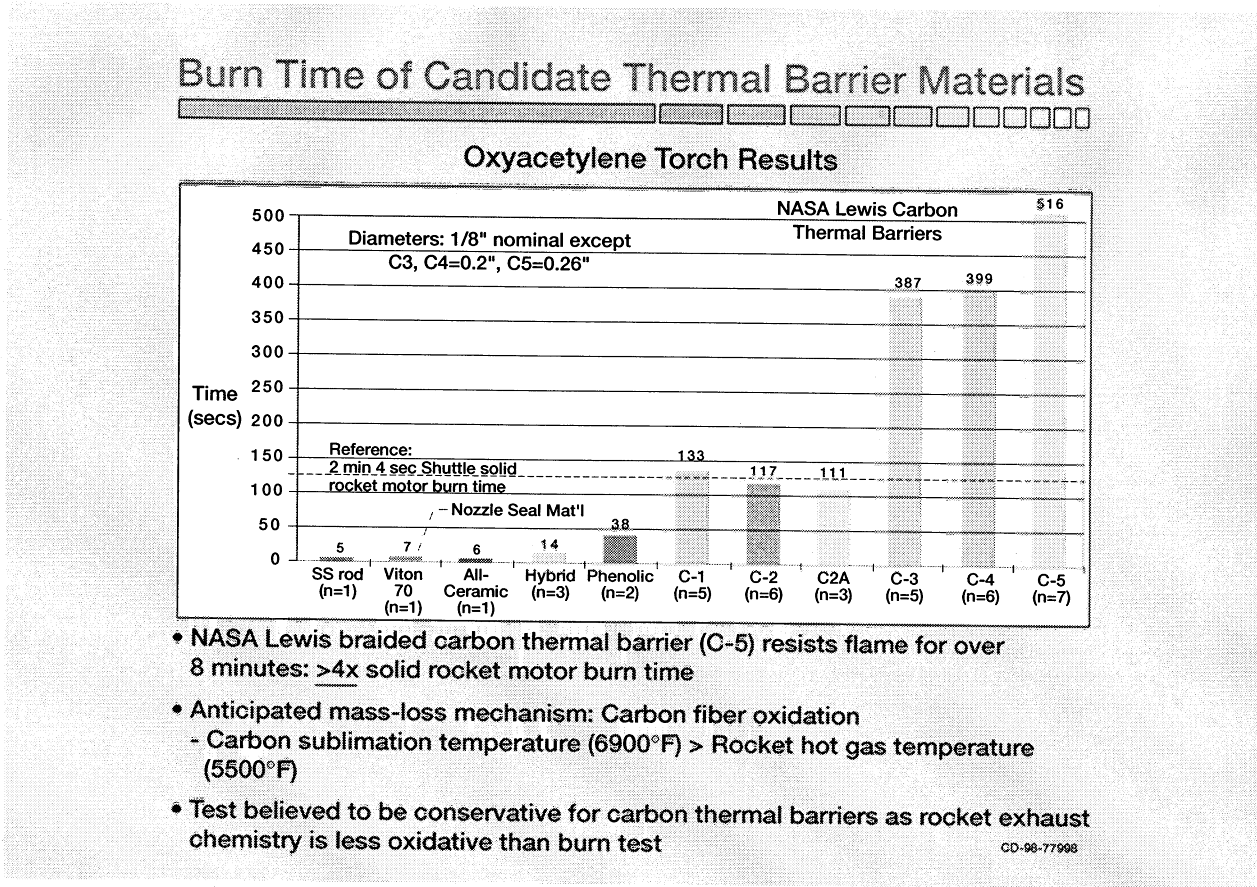
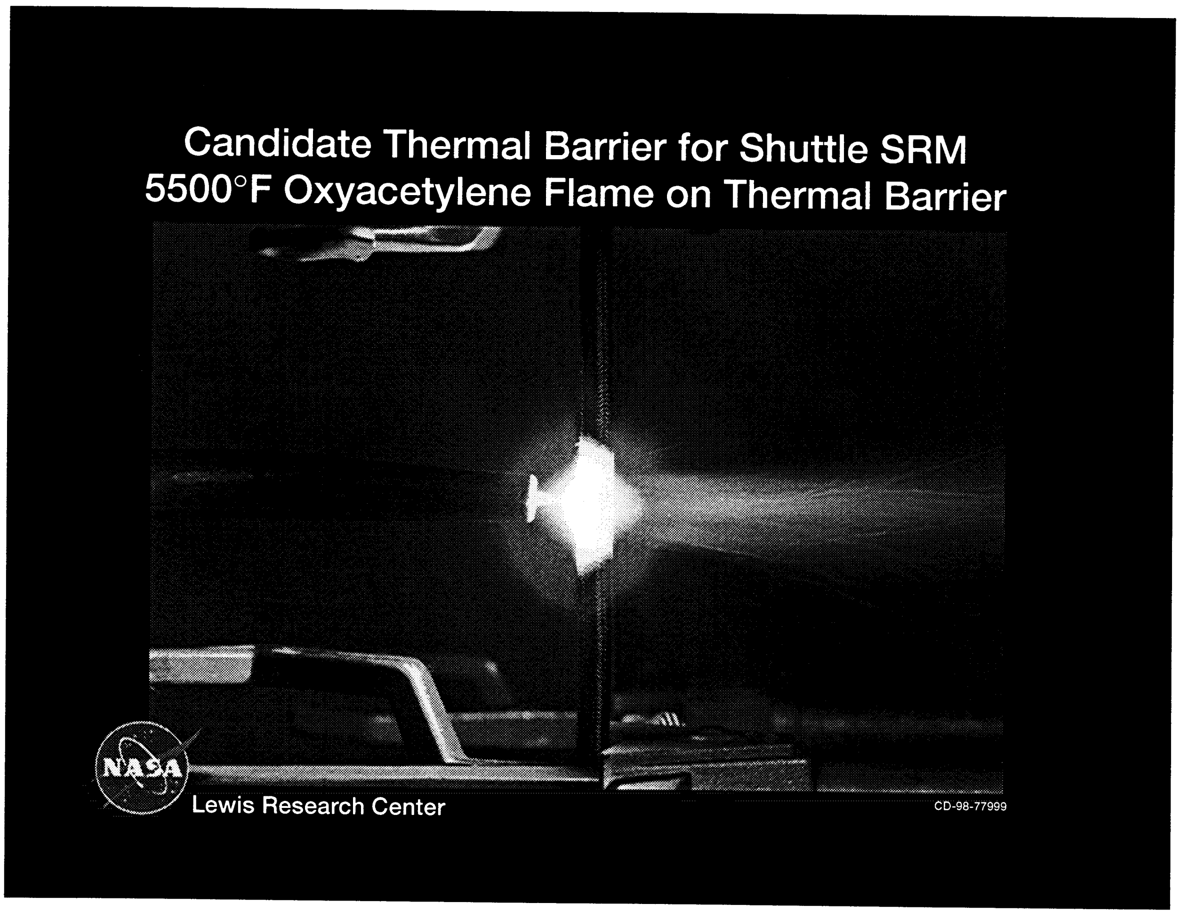
This chart shows a schematic of a simple fixture that we came up with to screen different thermal barrier materials and designs. Candidate materials are placed into the flame of an oxyacetylene torch and the amount of time to completely burn through them is measured. The gas mixture of the flame is adjusted until a neutral flame is formed, and the thermal barrier materials are placed in the hottest part of the flame at the tip of the inner cone where temperatures reach 5500°F. The fixture has a carriage, which can hold both a calibration specimen and a test specimen. The calibration specimen is used to make sure that the flame is adjusted properly and that the specimens are consistently in the hottest part of the flame from test to test. Once it is shown that the flame is adjusted correctly on the calibration specimen, the test specimen is slid into the flame and the amount of time to completely burn through the specimen is measured.

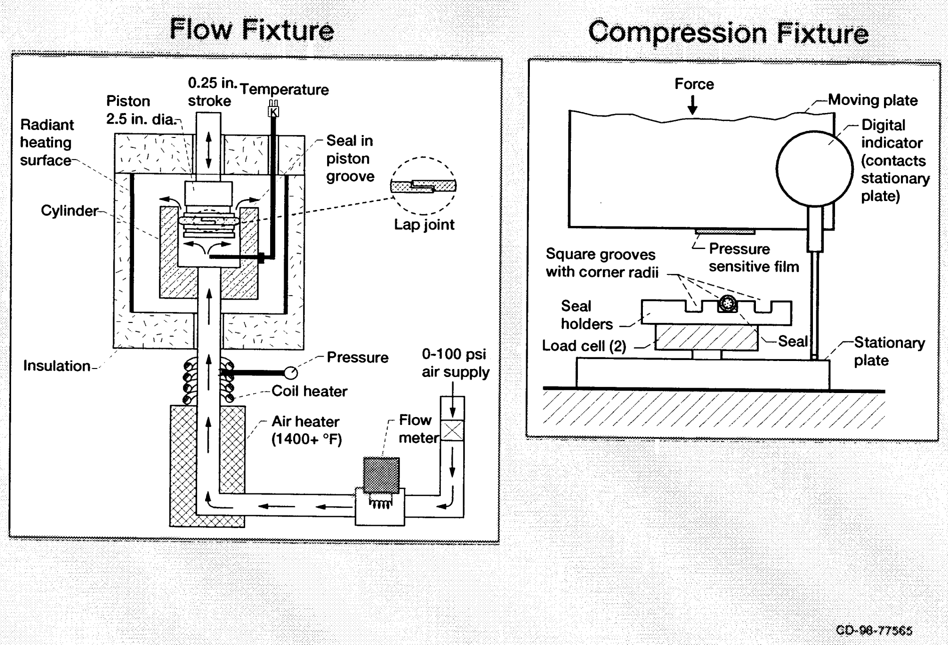


This chart shows the results of burn tests performed on candidate thermal barrier materials and designs. It shows the amount of time that it took for the oxyacetylene torch to cut completely through the different materials. All of the specimens were 1/8" in diameter except the Carbon-3 (C-3) and Carbon-4 (C-4) designs, which were 0.2" in diameter, and the Carbon-5 (C-5) design that was 0.26" in diameter. The first specimen that was tested was an 1/8" diameter stainless steel rod to get a reference point on how hot the flame was. This rod was cut through in only 5 seconds, and the metal was actually melted where the flame cut through. Next, an 1/8" diameter Viton O-ring, the same material used for the O-rings in the rocket nozzle, was tested. It was cut through in about 7 seconds. Two braided rope seal designs were then tested. Like all the other braided rope seal designs, including the thermal barriers, these seals are made up of a core of uniaxial fibers over which layers of sheath material are braided. The all-ceramic design is made up entirely of ceramic fibers, while the hybrid design has a core of ceramic fibers and a sheath braided out of superalloy wires. These designs lasted an average of 6 and 14 seconds, respectively, in the flame. Next, a phenolic thermal barrier was tested that was braided out of phenolic fibers similar to the material used as insulation in the rockets. This material lasted an average of 38 seconds in the flame. Finally, we get to the carbon thermal barriers. The three 1/8" diameter designs all lasted about 2 minutes in the oxyacetylene flame. As a point of comparison again, the Shuttle solid rocket motor burn time of 2 minutes 4 seconds is indicated in the figure. Moving up in diameter, the C-3 and C-4 designs with a diameter of 0.2" lasted about six and a half minutes in the flame. Finally, the 0.26" diameter C-5 design withstood the flame for over 8 minutes, more than four times the solid rocket motor burn time. Thus, the C-5 thermal barrier can endure 5500°F gases for over four times longer than the solid rockets would actually be in operation. After the carbon thermal barriers were removed from the flame, the areas where the flame cut through them were just as soft and flexible as before they were tested. There were no signs of melting, charring, or embrittlement. We believe that the carbon fibers are actually being oxidized as they are cut through. Carbon fibers oxidize at temperatures above 600 to 900°F depending on the type of fiber. At the other end of the spectrum, the sublimation temperature of carbon is at 6900°F. Therefore, the temperature of the rocket and the oxyacetylene torch at 5500°F is hot enough for oxidation of the fibers but not hot enough for sublimation to occur. Also, these tests are probably conservative in terms of burn resistance because they were performed in an oxidizing ambient atmosphere, whereas the rocket exhaust chemistry is not as oxidative. It is possible that the carbon thermal barriers could be exposed to rocket exhaust in the Shuttle solid rocket motors and not even be effected.



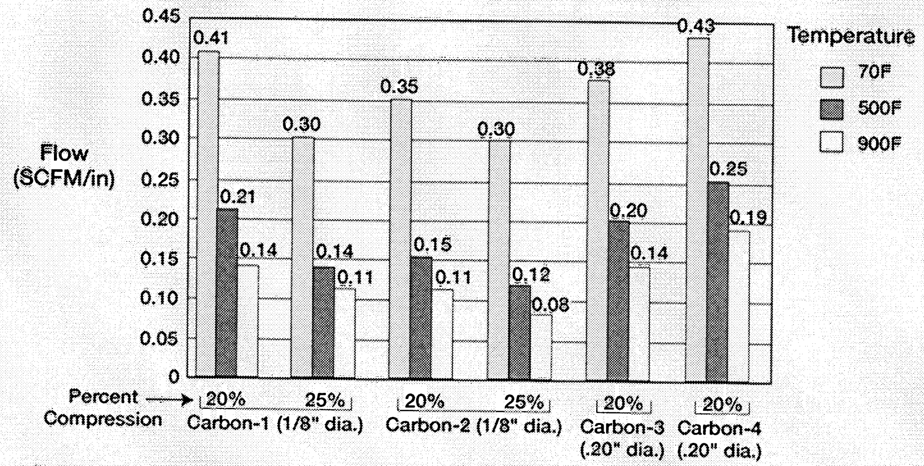
This chart shows the thermal barrier while it is being exposed to the oxyacetylene torch. An incandescent fireball can be seen around the thermal barrier, which is positioned at the tip of the inner cone of the flame. In actuality, this fireball is too bright to look at with the naked eye, and welding glasses must be used to view the test. However, this image was taken with a digital camera that filtered out the brightness of the flame.

Test Fixture Schematics



The next two schematics are for two more of our test fixtures, the flow fixture and the compression fixture. In the flow fixture seals and thermal barriers are installed into a groove in a piston which is then lowered into a cylinder. Hot, pressurized air is forced into the bottom of the cylinder, and the amount of flow past the seal is measured at different pressures and temperatures. The seals and thermal barriers are installed in the piston by forming a lap joint between their two ends, and the joint is oriented to minimize leakage through the two ends. The other figure shows the compression fixture which is used to determine the loading characteristics of different seal and thermal barrier designs. Seals are inserted into a machined groove, and an opposing plate is moved so that the seal is compressed against this plate. Load versus displacement curves can then be generated. When the seals are unloaded, the amount of resiliency and permanent deformation are measured. The contact area of the seals is also measured by using pressure sensitive film. The film changes color when it is loaded, and the contact area is measured as the area of the film that changed color when the seal was pressed against it.

Thermal Barrier Flow Results



- Flow resistance increases with higher compression.
- Mass flow less at higher temperatures because viscosity is higher.
- C-4 (larger diameter core fibers) results in roughly 20% greater flow than C-3.
- Braided thermal barrier exhibits satisfactory flow:
  - Low enough to drop temperature through thermal barrier
  - High enough to permit primary/secondary O-ring leak check without "false-positives"

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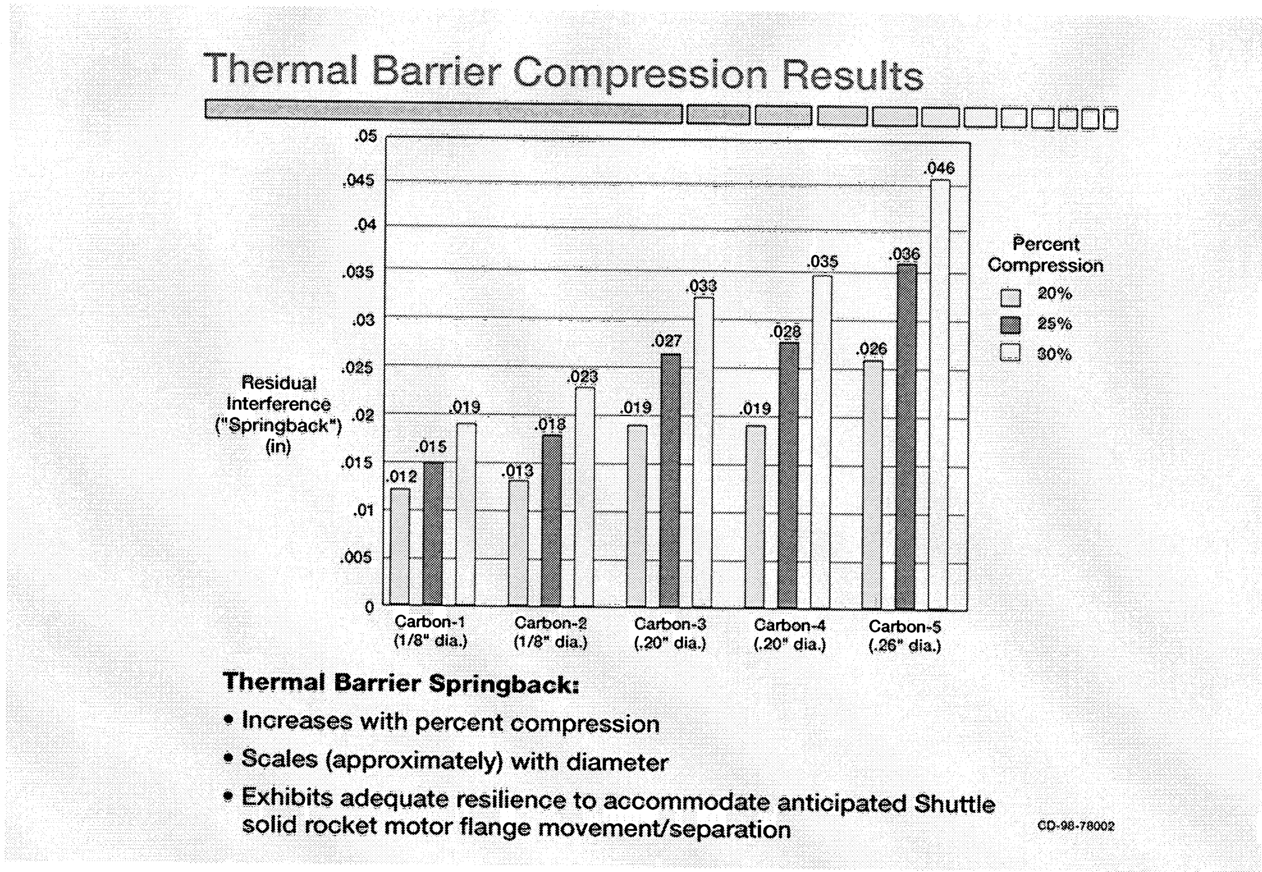
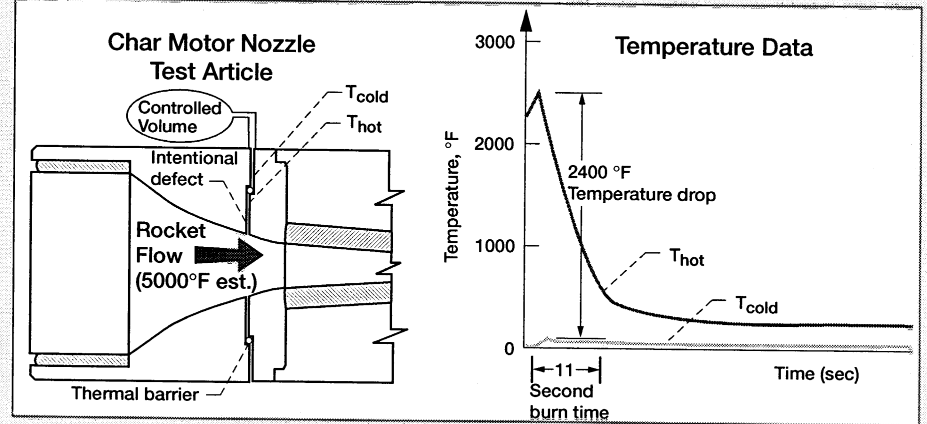
This chart shows the results of flow tests performed on the different thermal barrier designs. The figure shows flow rates in SCFM/in at a pressure of 60 psid at different compression levels on the thermal barriers. It can be seen that flow resistance increases with higher levels of compression. When the amount of linear compression was increased from 20% to 25% in the Carbon-1 and Carbon-2 designs, flow rates past the thermal barriers decreased. Also shown in this figure is that mass flow decreased as the temperature increased. At higher temperatures, air viscosity increases, so the amount of flow past the thermal barriers decreases. Overall, the braided thermal barriers restrict flow enough to drop temperatures across them, which will be elaborated on shortly, but are permeable enough to permit leak checks of the primary/secondary O-ring system without any false-positives.



This figure shows results of compression tests performed on the different thermal barrier designs. For each thermal barrier design, the amount of residual interference, or springback, is shown at 20%, 25%, and 30% linear compression. To define springback, use the 20% compression test of Carbon-3 as an example. A 20% compression of this 0.2" diameter thermal barrier means that it is compressed 0.040". For this amount of compression, the thermal barrier "springs back" and recovers 0.019". Thus, there is some permanent set for the thermal barriers when they are compressed. As shown in the figure, the amount of springback increases with percent compression and scales approximately with diameter. Overall, the thermal barriers exhibit adequate resilience to accommodate anticipated movements and separation of the Shuttle solid rocket motor flanges.

Thiokol Char Motor Test Results – Controlled Volume



- Char motor test with intentional joint defect allows hot rocket gas to impinge on thermal barrier for temperature and material performance evaluation
- Results: Thermal barrier provided huge temperature drop:
  - 2500+°F hot side; 100°F cold side: 2400°F delta T through diameter (Generation I 0.125" dia. thermal barrier). Larger diameters planned
  - No apparent burning of thermal barrier
- Char motor tests qualify thermal barrier for subsequent detailed test/evaluation

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Thiokol, who NASA Lewis is working with to develop the thermal barrier, has performed tests in a subscale rocket "char motor" in which the thermal barrier is exposed to hot rocket gases. The figure on the left shows how the hot gases would move through an intentional joint defect and impinge on the thermal barrier. Pressures and temperatures were measured both upstream and downstream of the thermal barrier. The plot on the right shows temperature traces from a test performed with an 1/8" diameter thermal barrier in the char motor. The plot shows that temperatures on the hot side of the thermal barrier reached 2500°F for an eleven-second rocket firing, while cold side temperatures only reached about 100°F for a 2400°F temperature drop across the thermal barrier. When the specimen was removed from the char motor, it was in excellent condition with no apparent burning or charring. This test qualified the thermal barrier for subsequent detailed testing and evaluation.

Thermal Barrier Development Program – NASA LeRC

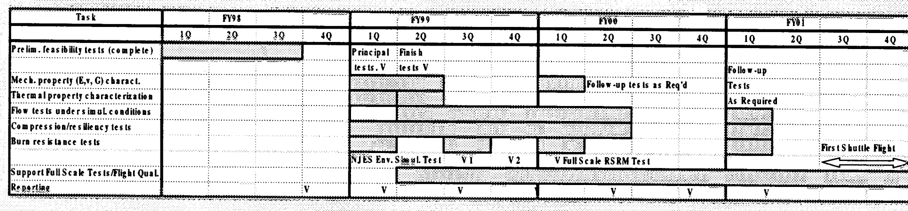
Objective:

Assist Thiokol in maturing the NASA Lewis thermal barrier to RSRM operational status.

Approach:

- Assist in defining optimum thermal barrier config. (architecture, joining, etc.)
- Characterize thermal barrier thermal (Cp, k) properties & mech (E, ϖ, G) properties
- Perform flow, burn-resistance, and compression ("spring-back") tests of the thermal barrier under various conditions
- Consult with Thiokol for preliminary /detailed design phases and for Full Scale RSRM rocket tests, flight qualification, and introduction to service.

Program:



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This chart summarizes the thermal barrier development program in which NASA Lewis will assist Thiokol in maturing the thermal barrier to RSRM operational status. Over the next several years, NASA Lewis plans to assist in defining an optimum thermal barrier configuration by performing additional testing and material characterization on the thermal barrier. Thermal properties, such as heat capacity, conductivity, and coefficient of thermal expansion, and mechanical properties, including elastic modulus, Poisson's ratio, and ultimate strength, will all be determined. Additional flow, resiliency, and burn-resistance tests are also planned. NASA Lewis plans to consult with Thiokol throughout the design phases and through flight qualification and full scale RSRM rocket tests until the thermal barrier is ready for introduction to service. The figure at the bottom shows the proposed development schedule, including a full scale RSRM test and the first planned Shuttle flight using the thermal barrier.

Video:

A video was shown to highlight the burn resistance of the thermal barrier. After showing a 1/8" diameter stainless steel rod being cut through in only 5 seconds, portions of burn tests on Carbon-1 and Carbon-3 thermal barriers were shown. The 1/8" diameter Carbon-1 thermal barrier was cut through in about 2 minutes 30 seconds, while the 0.2" diameter Carbon-3 design took over 6 minutes to be cut through by an oxyacetylene torch.

Summary and Conclusions

- NASA Lewis 0.26" diameter thermal barrier resists 5500°F flame for 8 min. 36 sec. before burn-through:
  - Lasted ≥4X Shuttle solid rocket motor burn-time.
  - Anticipated mass-loss mechanism: Carbon oxidation
- Thermal barrier remains flexible even in hottest burn zone.
- Braided thermal barrier
  - Blocks hot gas flow but permits primary/secondary O-ring leak check
  - Exhibits adequate resilience to accommodate SRM flange movement
- Char motor tests under simulated rocket environment showed thermal barrier provided 2400°F temperature drop through 1/8" diameter thermal barrier.

Thermal barrier feasibility established qualifying it for comprehensive evaluation

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In summary, the NASA Lewis thermal barrier has shown great promise for use in RSRM nozzle joint locations. The 0.26" diameter thermal barrier has shown the ability to resist the 5500°F flame of an oxyacetylene torch for over eight and a half minutes, more than four times the burn time of the Shuttle solid rocket motors. It is believed that the actual mass-loss mechanism involved in this process is oxidation of the carbon fibers that make up the thermal barrier. After the thermal barrier has been cut through, it remains flexible even in the hottest burn zone with no signs of charring or melting. The thermal barrier blocks hot gas flow well enough to drop gas temperatures across the barrier but is still permeable enough to permit a leak check of the primary and secondary O-rings in the nozzle-to-case joint of the solid rocket motors. It also exhibits adequate resilience to accommodate RSRM flange movements. Char motor tests performed by Thiokol under a simulated rocket environment showed that an 1/8" diameter thermal barrier provided a 2400°F temperature drop across the thermal barrier. In conclusion, thermal barrier feasibility has been established qualifying it for comprehensive evaluation.