7/13/90
E5564

NASA Contractor Report 185257
AIAA-90-2116

Rocket Combustion Chamber Life-Enhancing Design Concepts

Richard J. Quentmeyer
Sverdrup Technology, Inc.
Lewis Research Center Group
Brook Park, Ohio

July 1990

Prepared for
Lewis Research Center
Under Contract NAS3-25266

NASA
National Aeronautics and
Space Administration

ROCKET COMBUSTION CHAMBER LIFE-ENHANCING DESIGN CONCEPTS

R.J. Quentmeyer
Sverdrup Technology, Inc.
Lewis Research Center Group
Brook Park, Ohio 44142

ABSTRACT

NASA continues to pursue technologies which can lead to an increase in life and reduce the costs of fabrication of the Space Shuttle Main Engine. The joint NASA/Air Force Advanced Launch System Program has set its prime objectives to be high reliability and low cost for their new advanced booster engine. In order to meet these objectives, NASA will utilize the results of several ongoing programs to provide the required technologies.

This paper presents an overview of those programs which address life-enhancing design concepts for the combustion chamber. Seven different design concepts, which reduce the thermal strain and/or increase the material strength of the combustion-chamber-liner wall are discussed. Subscale rocket test results are presented, where available, for life-enhancing design concepts. Two techniques for reducing chamber fabrication costs are discussed, as well as issues relating to hydrocarbon-fuels/combustion-chamber-liner materials compatibility.

E-5564

INTRODUCTION

The Space Shuttle Main Engine (SSME) was the first high performance rocket engine designed to be reusable. However, the actual life of the main combustion chamber (MCC) is considerably less than the original design life due to cracks which form in the MCC liner wall. The cracks are the result of a large, thermally-induced, plastic strain, which causes the wall between the cooling passages and the hot-gas side to deform and progressively thin with each thermal cycle. This phenomenon is known as thermal ratcheting and after numerous thermal cycles, cracks develop. Figure 1 shows an example of this type of failure.

Over the past 20 years, NASA has sponsored many programs whose objectives were to better understand the failure mechanism in combustion chamber liners, to develop models which could be used to predict chamber life, to screen for candidate combustion-chamber-liner materials, and to evaluate the merits of new design concepts for improving chamber life (refs. 1 to 12).

Recently, the joint NASA/Air Force Advanced Launch System (ALS) program has increased the emphasis on high reliability and low cost for their proposed booster engine design. This creates the need to explore new fabrication techniques to reduce costs, as well as the need to create an engine design that has an inherently long life.

Although the current ALS booster engine design studies are focused on a hydrogen/oxygen propellant combination, the original engine design called for a hydrocarbon-fuel/liquid-oxygen propellant combination using the hydrocarbon fuel as the coolant. However, heat transfer investigations using hydrocarbon fuels indicated that copper materials were subject to corrosion when exposed to the fuel (refs. 13 to 15). This prompted studies to explore the use of hydrogen as a second fuel on the vehicle for combustion chamber cooling, or to consider liquid oxygen as the coolant (refs. 16 to 18). Thus, no longer was life improvement a question, but rather finding a way to prevent life reduction due to fuel/materials compatibility issues.

In a continuing effort to make improvements in the SSME and to create long-life, low-cost designs for future high-pressure rocket engines, NASA and the Air Force have funded several programs to provide the required technologies. This paper presents an overview of seven design concepts which address the issue of combustion-chamber-liner life-enhancement, although the issues pertaining to hydrocarbon-fuels/materials compatibility and low cost fabrication techniques are also discussed. Experimental test results from the NASA Lewis Research Center subscale rocket engine test apparatus, which has been used to evaluate some of the life-enhancing design concepts reported herein, are also presented.

SUBSCALE ROCKET ENGINE TEST APPARATUS

In order to screen materials, provide data for model development and to evaluate new design concepts, a low cost, subscale rocket engine test apparatus was designed at NASA's Lewis Research Center (LeRC). Figure 2 shows a schematic of this apparatus. The apparatus consists of an injector, an outer cylindrical chamber which serves as the test section, and a water-cooled centerbody which forms the chamber throat. Liquid oxygen and gaseous hydrogen are used as propellants, creating a throat heat flux of approximately 97.1 mW/m^2 (33 Btu/in.^{2}/sec) at a chamber pressure of 4.14 MPa (600 psia). However, plans are in progress to increase the heat flux level to 250 mW/m^2 (85 Btu/in^2-sec) by increasing the chamber pressure to 13.8 MPa (2000 psia). The chamber is separately cooled with liquid hydrogen and is ideally suited to conduct cyclic life tests, because it can be thermally cycled from 28 K (50.4 °R) to a throat wall temperature of 800 K (1440 °R) and back to liquid hydrogen temperature in just 3.5 sec. Since the chamber is not contoured, numerous design concepts can be evaluated for relatively low fabrication costs.

LIFE-ENHANCING DESIGN CONCEPTS

The conventional method of making a combustion chamber liner for high chamber pressure operation, such as the SSME, is to machine cooling passages into a contoured spun liner made of a high-strength copper alloy. The liner is closed out with a nickel jacket to which a high-strength structural jacket is attached to contain the chamber pressure and transmit the thrust load. Although this design was state-of-the-art at the time of the development of the SSME, the stiff liner closeout prevents the inner chamber wall from expanding while the engine is firing. This results in a high, thermally-induced plastic strain which causes cracks to form in the cooling passage/hot-gas-side wall after repeated thermal cycles.

The following describes various design concepts which could increase the life of the SSME MCC, or that of a future high-pressure engine, either by reducing the thermal strain, or by increasing the strength of the liner material.

Thermal Barrier Coatings

One method of reducing, or eliminating, the large plastic strain in the hot-gas-side wall is to apply a thermal barrier coating (TBC), such as zirconium-oxide (ZrO_2). The coating performs two functions which reduce the thermal strain in the wall. If the coating is allowed to operate at a temperature of 1944 K (3500 °R) on the hot-gas side of the wall, the heat flux is reduced nearly 50 percent. Also, since the coating is an insulator, there is a large temperature drop across the coating, resulting in a much lower metal wall temperature. The combined effect results in a dramatic reduction in the thermal strain and an increase in material strength.

The use of TBC's on rocket combustion chambers has never been widely accepted due to the tendency of the coatings to flake off. However, a method of applying a TBC was developed (ref. 19), which has proven not only to be durable, but to markedly increase chamber life. In this case, the chamber is fabricated from the inside out. The process consists of plasma spraying the TBC onto a mandrel, applying a thin bond coat, such as nickel-chromium (NiCr), to the TBC, and then electroforming the combustion chamber liner around the coating. This results in a smooth, durable coating, free of residual compressive stresses and eliminates the potential oxidation of the metallic substrate during the application of the coating, which is detrimental to the adherence of the coating.

This concept has been evaluated on the Lewis subscale rocket engine test apparatus (refs. 20 and 21). Figures 3 and 4 show post-test wall cross-sections in the throat region of two chambers having the same cooling passage geometry and operated at the same conditions. One chamber had a liner fabricated from Amzirc (Cu,0.15%Zr), a high strength copper alloy, which is considerably stronger than copper, while the other chamber had a liner made from electrodeposited copper (ED Cu). The uncoated Amzirc chamber, shown in figure 3, developed a crack in the liner wall after 393 thermal cycles. Deformation and thinning of the cooling passage wall is readily apparent. The chamber with the ED copper liner, shown in figure 4, had a 0.076 mm (0.003 in.) thick ZrO_2 coating with a 0.025 mm (0.001 in.) thick NiCr bond coat applied by the technique described above. The cooling passage wall shows no sign of damage after 1450 thermal cycles. This method of producing a long-life combustion chamber shows considerable promise.

Tungsten-Reinforced Chamber Liner

Subscale rocket tests have demonstrated that a tungsten-reinforced combustion chamber liner can also be beneficial in improving chamber life (ref. 22). This design concept does not reduce the strain level in the liner wall, but rather greatly increases the strength of the material and prevents the deformation and thinning of the wall.

The fabrication of this type of chamber is also accomplished from the inside out. The sequence consists of arc spraying the copper alloy onto a steel mandrel to the desired wall thickness and then circumferentially machining a low angle helix into the surface. A continuous tungsten wire is wrapped onto the helix and another layer of the copper alloy is applied over the tungsten wire to a thickness sufficient to machine in the cooling passages. The assembly is hot isostatic pressed (HIP'ed) after each application of the copper alloy in order to densify the material. The fabrication of the remainder of the chamber is completed in the conventional manner.

Figure 5 shows a post-test longitudinal cross-section of a chamber liner with 0.020 cm (0.008 in.) diameter tungsten wires imbedded in a 0.089 cm (0.035 in.) thick copper wall. The wire spacing is one wire diameter and the resulting thermal conductivity of the composite is only 10 percent lower than the thermal conductivity of oxygen-free, high-conductivity (OFHC) copper. However, the rupture strength of the composite is 80 percent higher than the rupture strength of NARloy-Z (Cu,3.5%Ag,0.5%Zr) at 867 K (1560 °R), which is the copper alloy used for the SSME MCC. Figure 6 shows a cross-section of a chamber liner in the throat plane after 400 thermal cycles. No damage occurred to the chamber, i.e., no deformation or thinning of the wall between the cooling passages and the hot-gas side.

Hot-Gas-Side Slots

Another potential method of reducing the strain in the hot-gas wall is to provide a slot between each cooling passage to take up the thermal expansion. Figures 7 and 8 show an example of this design concept. The liner is machined in the conventional manner, but prior to closing out the cooling passages, a hole is drilled through each cooling passage rib at each end of the chamber (fig. 7). Slots are cut through the ribs by electrical discharge machining (EDM) using a 0.010 cm (0.004 in.) diameter wire, resulting in a 0.013 cm (0.005 in.) wide slot. The slots could also be selectively located, i.e., in the throat region only. This design concept has yet to be evaluated in hot-fire tests. Other methods of providing hot-gas-side slots are also being investigated.

Tubular-Bundle

The tubular-bundle design concept is not new in that most of the expendable, low-chamber-pressure rocket engines, past and present, incorporate a design in which thin stainless steel tubes are brazed together to make up the cooling passages of the combustion chamber. However, the use of stainless tubes for high-pressure combustion chambers is not practical, as the conductivity is too low and the tubes would have to be very thin to keep the operating temperature within the material limits. As a result, the tubes would not be able to carry the coolant pressure load.

Although copper-alloy tubes have not been used for the liner of a high-pressure combustion chamber, their use may offer several advantages. The tube bundle is more strain tolerant, because the crown of the tube can expand in order to relieve the thermal strain. Due to the increase in the hot-gas-side surface area over that of a machined cylindrical liner, more heat can be picked up by the coolant, which can increase the power of the turbine drives for the pumps (ref. 23). The tubes can also be mass-produced, which reduces manufacturing costs. Other advantages can be achieved if the tubes are joined together by electroforming (ref. 24). A small gap can be left between each channel, emulating the hot-gas-side slot concept, in order to provide for more expansion. This can not be done with brazing, because tubes must be tightly stacked to form good braze joints. The manifolds can also be formed, or attached, by electroforming.

The unique feature of this fabrication concept is that no heat is required to assemble the components, thus preserving the material properties and the integrity of joints which may be sensitive to the high heat required for welding. Figure 9 shows a wall cross-section of a subscale copper-tube chamber liner fabricated by this technique. Although the example shown has round tubes, the tubes for a contoured rocket combustion chamber would be formed into a noncircular shape to accommodate the geometry change and to provide high-aspect-ratio (height/width) cooling passages.

High-Aspect-Ratio Cooling Passages

High-aspect-ratio (height/width) cooling passages have the potential to reduce the thermal strain in the wall and increase the material strength by substantially reducing the hot-gas-side wall temperature. The temperature reduction is achieved because of the increased fin effect resulting from the increase in the number of cooling passages. An analysis performed on the Lewis subscale rocket chamber showed that the wall temperature at the throat could be reduced from 777 K (1400 °R) to 444 K (800 °R) by increasing the number of cooling channels from 72, the baseline configuration, to 400, assuming the same coolant flow. Figure 10 shows the subscale chamber liner incorporating this design concept. This particular configuration has four bifurcations with 100, 200, and 400 channel sections. There are 400 channels located in the high heat flux region of the throat. Figure 11 is a cross-section of the 400 channel region. The channels and ribs are approximately 0.025 cm (0.01 in.) wide, having an aspect ratio of 6. Although this configuration has a 0.089 cm (0.035 in.) thick wall to compare data with the baseline configuration, the wall could be much thinner, which would further reduce the wall temperature and increase strength.

Another advantage of this concept is that the increased heat transfer provides more heat for the turbine drives on the pumps. Also, the pressure drop in the cooling passages can be reduced, because the coolant Mach number can be lowered with only a minor effect on the wall temperature.

Transpiration-Cooled Throat

The life-enhancing feature of the transpiration-cooled throat is that it can virtually eliminate the plastic strain encountered in the throat region of regeneratively-cooled, high-chamber-pressure combustion chambers. Figure 12 shows a schematic of this concept. Transpiration cooling is also not a new concept, but it has not been widely accepted because of the associated performance loss, the difficulty of fabricating the porous media and, in the case of hydrocarbon-cooled chambers, the potential to clog the porous media with carbon deposits. However, the use of platelets (described in the "Low-Cost

Fabrication Techniques" section) can reduce the complexities and costs associated with manufacturing the porous media section of the liner. And due to the ability to accurately meter the fuel cooling with the platelet design, the performance losses can be minimized. Another advantage associated with transpiration cooling is that a high-velocity coolant is not required to cool the throat region, which reduces the pressure drop and the required fuel pump discharge pressure.

Low-Stiffness Closeout

Another life-enhancing design concept is to provide a low-stiffness closeout for the combustion chamber liner, which allows the hot-gas-side wall to expand, reducing the thermal strain in the wall. This concept was proposed several years ago, incorporating a fiberglass overwrap around the copper liner (ref. 25). The fiberglass was compliant enough to expand with the liner, but still had sufficient strength to contain the high chamber pressure. Figure 13 shows a chamber using this concept. Analytical and experimental results showed that an increase in chamber life, when compared to an identical chamber with a conventional closeout, could be achieved. However, ruptures in the copper closeout on the back-side of the liner resulted in hydrogen leaks, indicating that the stiffness of the fiberglass overwrap may have been too low.

Figure 14 illustrates the concept proposed by the authors of reference 26. The design consists of a conventionally machined, copper closed-out liner with compliant, or low-stiffness, layers between the liner and the structural jacket. In this case, a layer of sintered aluminum alloy backed by a polytetrafluoroethylene (PTFE) insulating layer was used for the intermediate layers. A structural analysis of this design concept indicated a life-enhancement factor of three over an identical chamber having the conventional electroformed nickel closeout between the liner and the structural jacket.

HYDROCARBON-FUELS/COMBUSTION-CHAMBER-LINER MATERIALS COMPATIBILITY

Another issue that must be addressed when designing a rocket combustion chamber for long life is the compatibility of the coolant, typically the fuel, with the chamber liner materials. One of the ALS booster engine concepts was to use a hydrocarbon-fuel/liquid-oxygen propellant combination in order to take advantage of the resulting vehicle high mass fraction during the near sea-level portion of the flight. However, previous heat transfer investigations indicated that copper and copper-base alloys were severely attacked by hydrocarbon fuels. As a result, the use of hydrocarbon fuels to cool copper chamber liners was in question.

In order to resolve this issue, an effort was initiated to determine the corrosive interaction between hydrocarbon fuels and candidate combustion chamber liner materials, and to develop and evaluate protective measures. As a result of this program, it was determined that trace amounts of sulfur in the fuel caused the reaction with copper cooling passages, and not the basic fuel (refs. 27 and 28). Figure 15 shows the bottom of a copper alloy cooling channel cooled by RP-1 contaminated with 50 ppm of mercaptan sulfur. The sulfur reacted with the copper to form copper sulfide, which formed a rough, coolant-flow-restricting and heat-transfer-reducing deposit on the cooling channel wall. The severe reaction of sulfur with copper would be very detrimental to the life of a rocket combustion chamber liner constructed of copper.

However, no reaction was observed between hydrocarbon fuels and the copper alloys evaluated in this program if the sulfur content in the fuel was below 1 ppm. Also, a gold coating applied to the cooling channel was found to prevent the reaction of sulfur contaminated fuels with copper (refs. 29 and 30). However, applying the gold coatings to the cooling channels would result in increased fabrication costs. Thus, the use of hydrocarbon fuels for cooling in a high-pressure rocket combustion chamber would require that the fuel be virtually sulfur-free.

LOW-COST FABRICATION TECHNIQUES

The need to reduce the fabrication costs of the SSME and the emphasis of low cost for the ALS booster engine has resulted in new fabrication techniques to meet those objectives. Two of the techniques, the vacuum-plasma-sprayed (VPS) liner and the platelet-formed chamber liner are discussed here.

Vacuum-Plasma-Sprayed Liner

Two VPS concepts are presently being pursued to reduce fabrication costs of the SSME MCC. The two concepts, shown in figure 16, utilize the Vacuum-Plasma-Spray (VPS) technique to form the chamber liner.

In concept 1, the fabrication is performed from the inside out. The copper-alloy liner is formed by spraying NARloy-Z (Cu,3.5%Ag,0.5%Zr) onto a mandrel. Cooling channels are machined into the liner and then the channels are closed out by the VPS method. The structural jacket is formed in one continuous operation by using the VPS technique. After the fabrication is completed, the mandrel is removed. Presently, the structural jacket is fabricated in pieces then welded together.

The fabrication sequence in concept 2 is performed from the outside in. In this case, the process starts with a cast structural jacket to which the NARloy-Z is sprayed from the inside. The cooling channels are then machined on the inside diameter of the assembly. The channels are closed-out by VPS, after which the final machining of the chamber contour is performed from the inside. The unique feature of concept 2 is that the manifolds can be cast as an integral part of the structural jacket, which would greatly reduce the costs and risks associated with welding those components to the chamber.

Platelet-Formed Chamber Liner

The platelet-formed chamber liner has the same life-enhancing features described in the "High-Aspect-Ratio Cooling Passages" section. However, in this fabrication process, the high-aspect-ratio cooling passages can be made to almost any size at a much lower cost than conventional machining methods.

The chamber fabrication is accomplished by diffusion bonding flat, stacked platelets which have the proper cooling passage geometry already etched into the material. The flat panels are formed into contoured chamber segments on a die and then welded together to form the chamber. Figure 17 shows cross-sectional views of two different cooling passages geometries. Cooling passage aspect ratios as high as 15 have been formed using this fabrication process (ref. 31). Another advantage of this technique is that a very thin hot-gas-side wall can be fabricated with great accuracy, which would be very costly and entail high risks using conventional machining processes.

COMPARISON OF CONCEPTS

There is not enough experimental data to determine which of the seven life-enhancing concepts will be most beneficial, as only a few of the concepts have been evaluated in subscale rocket chamber tests. However, some comparisons of the concepts can be made.

The TBC and the transpiration-cooled throat concepts would probably provide the maximum reduction in thermal strain, resulting in the best life-enhancing capabilities. The TBC concept has already demonstrated its life-enhancing capability in subscale tests. Both of these concepts minimize the heat pickup in the coolant and are best suited for the gas generator cycle. The coating acts as an insulator and reduces the heat pickup by the coolant, which affects the power available for the turbine drives on the pumps. In transpiration cooling, the heat picked up by the coolant in the transpiration-cooled region is unavailable for the turbine drives as the coolant is dumped into the hot-gas stream. Conversely, the use of the tubular-bundle and high-aspect-ratio concepts maximizes the heat pickup by the coolant, which would be ideal for the expander cycle.

The tubular-bundle, made by the electroforming process, the hot-gas-side slots, and the low-stiffness closeout are all strain reducing designs, which allow the hot-gas-side wall to expand. However, the effect of the hot combustion-gases in the slots needs to be investigated.

The tungsten-reinforced chamber liner does not reduce the thermal strain, but rather greatly increases the strength of the wall material and prevents the deformation and thinning of the wall, as shown in subscale testing. The tungsten-reinforced liner concept could be combined with the high-aspect-ratio concept to produce low wall temperatures with greatly increased strength, provided that the optimum wall thickness for the high-aspect-ratio concept is not too thin to accommodate the tungsten wire.

The platelet-formed combustion chamber liner concept appears to have the most advantages - the ability to incorporate very high aspect ratio cooling passages, to allow for the manufacture of very accurate, thin walls, and potential for low fabrication costs. Likewise, the VPS concept also has the potential to greatly reduce fabrication costs. However, in the case of the SSME MCC sprayed liner, the resulting thermal strain will not be less than on the present MCC, unless one of the other life-enhancing design concepts is incorporated into the fabrication.

Also, the issue of materials compatibility must be addressed if hydrocarbon fuels are to be used to cool combustion chamber liners made of copper alloys.

CONCLUSIONS

The TBC and transpiration-cooled throat concepts appear to have the best life-enhancing capabilities, but are best suited for the gas generator cycle.

The tubular-bundle and high-aspect-ratio concepts appear to have lesser life-enhancing capabilities, but are best for the expander cycle.

The tungsten-reinforced liner concept, while not a thermal strain-reducing concept, has demonstrated in subscale tests that it has good life-enhancing capabilities.

The platelet-formed and vacuum-plasma-spray-formed chamber concepts have the potential to substantially reduce fabrication costs.

If hydrocarbon fuels are used to cool combustion chamber liners fabricated from copper alloys, the fuels must be virtually sulfur-free, or the cooling passages must have a protective coating, such as gold, to prevent corrosion by trace amounts of sulfur in the fuel.

REFERENCES

1. Conway, J.B.; Stentz, R.H.; and Berling, J.T.: High Temperature, Low-Cycle Fatigue of Copper-Base Alloys in Argon. Part I: Preliminary Results for 12 Alloys at 1000 °F (538 °C). (Mar-Test, Inc., NASA Contract NAS3-16753) NASA CR-121259, 1973.

2. Esposito, J.J.; and Zabora, R.F.: Thrust Chamber Life Prediction, Volume I, Mechanical and Physical Properties of High Performance Rocket Nozzle Materials. (Boeing Aerospace Co., NASA Contract NAS3-17838) NASA CR-134806, 1975.

3. Armstrong, W.H.; and Brogren, E.W.: Thrust Chamber Life Prediction, Volume II, Plug Nozzle Centerbody and Cylinder Life Analysis. (Boeing Aerospace Co., NASA Contract NAS3-17838) NASA CR-134822, 1975.

4. Armstrong, W.H.; and Brogren, E.W.: Thrust Chamber Life Prediction, Volume III, Fatigue Life Parametric Study. (Boeing Aerospace Co., NASA Contract NAS3-17838) NASA CR-134823, 1975.

5. Armstrong, W.H.; and Brogren, E.W.: 3-D Thrust Chamber Life Prediction. (Boeing Aerospace Co., NASA Contract NAS3-19717) NASA CR-134979, 1976.

6. Hannum, N.P.; Kasper, H.J.; and Pavli, A.J.: Experimental and Theoretical Investigation of Fatigue Life in Reusable Rocket Thrust Chambers, AIAA Paper 76-685, July 1976. (Also, NASA TM X-73413. 1976).

7. Quentmeyer, R.J.: Experimental Fatigue Life Investigation of Cylindrical Thrust Chambers, AIAA Paper 77-893, July 1977. (Also, NASA TM X-73665, 1977).

8. Hannum, N.P.; Quentmeyer, R.J.; and Kasper, H.J.: Some Effects of Cyclic Induced Deformation in Rocket Thrust Chambers. (Conference on Advanced Technology for Future Space Systems) AIAA, 1979, pp. 290-299. (Also, NASA TM-79112, 1979).

9. Armstrong, W.H.: Structural Analysis of Cylindrical Thrust Chambers, Volume I. (LMSC-HREC-TR-D568827-VOL-1 Lockheed Missiles & Space Company, Inc., NASA Contract NAS3-21361) NASA CR-159522, 1979.

10. Armstrong, W.H.: Structural Analysis of Cylindrical Thrust Chambers, Volume II. (LMSC-HREC-TR-D698400-VOL-2 Lockheed Missiles & Space Company, Inc., NASA Contract NAS3-21953) NASA CR-165241, 1981.

11. Armstrong, W.H.: Structural Analysis of Cylindrical Thrust Chambers, Volume III. (LMSC-HREC-TR-D698440-VOL-3 Lockheed Missiles & Space Company, Inc., NASA Contract NAS3-21953) NASA CR-165242, 1981.

12. Cook, R.T.; Fryk, E.E.; and Newell, J.F.: SSME Main Combustion Chamber Life Prediction. (RI-RD83-150 Rocketdyne Division, Rockwell International, NASA Contract NAS3-23256) NASA CR-168215.

13. Roback, R.; Szetela, E.J.; and Spadaccini, L.: Deposit Formation in Hydrocarbon Fuels. (United Technologies Research Center, NASA Contract NAS3-22277) NASA CR-165405, 1981.

14. Giovanetti, A.J.; Spadaccini, L.J.; and Szetela, E.J.: Deposit Formation and Heat Transfer in Hydrocarbon Rocket Fuels. (United Technologies Corp., NASA Contract NAS3-23344) NASA CR-168277, 1983.

15. Cook, R.T.: Methane Heat Transfer Investigation. (Rockwell International Corp., NASA Contract NAS8-34977) NASA CR-171199, 1985.

16. Price, H.G.; and Masters, P.A.: Liquid Oxygen Cooling of High Pressure LOX/Hydrocarbon Rocket Thrust Chambers. NASA TM-88805, 1986.

17. Armstrong, E.S.: Liquid Oxygen Cooling of Hydrocarbon Fueled Rocket Thrust Chambers. AIAA Paper 89-2739, July 1989.

18. Armstrong, E.S.: Cooling of Rocket Thrust Chambers with Liquid Oxygen. To be published. AIAA Paper 90-2120, July 1990.

19. Hammer, S.; and Czacka, Z.: Development of Advanced Fabrication Techniques for Regeneratively Cooled Thrust Chambers by the Electroforming Process. (Camin Labs., Inc., NASA Contract NAS3-10304) NASA CR-72698, 1967.

20. Quentmeyer, R.J.; Kasper, H.J.; and Kazaroff, J.M.: Investigation of the Effect of Ceramic Coatings of Rocket Thrust Chamber Life. AIAA Paper 78-1034, July 1978. (Also, NASA TM-78892, 1978).

21. Quentmeyer, R.J.: Thrust Chamber Thermal Barrier Coating Techniques. Application of Advanced Material for Turbomachinery and Rocket Propulsion. AGARD-CP-449, Mar. 1989, pp. 18-1 to 18-10. (Also, NASA TM-100933, 1988).

22. Kazaroff, J.M.; and Jankovsky, R.S.: Cyclic Hot Firing Results of Tungsten Reinforced Copper Liner Thrust Chambers. NASA TM-4214, 1990.

23. Kacynski, K.J.: Three-Dimensional Heat Transfer Analysis for Advanced Tubular Rocket Thrust Chambers. To be published as AIAA Paper 90-2727, July 1990.

24. Kazaroff, J.M.; and Pavli, A.J.: Advanced Tube-Bundle Rocket Thrust Chamber. To be published as AIAA Paper 90-2726, July 1990.

25. Kasper, H.J.; and Notardonato, J.J.: Effect of Low-Stiffness Closeout Overwrap on Rocket Thrust-Chamber Life. NASA TP-1456, 1979.

26. Kumakawa, M.N.; Sumiyoshi, T.H.; and Watanabe, R.: Life Prediction of CIP Formed Thrust Chamber -- Cold Isostatic Pressing. IAF Paper 85-192, Oct. 1985.

27. Gage, M.L.: Hydrocarbon-Fuel/Combustion-Chamber-Liner Materials Compatibility, Interim Final Report. (Aerojet TechSystems Co., NASA Contract NAS3-25070) NASA CR-185203, 1990.

28. Rosenberg, S.D.; and Gage, M.L.: Compatibility of Hydrocarbon Fuels with Booster Engine Combustion Chamber Liners. AIAA Paper 88-3215, July 1988.

29. Rosenberg, S.D.; and Gage, M.L.: Corrosion Prevention in Copper Combustion Chamber Liners of Liquid Oxygen/Methane Booster Engines. To be published as AIAA Paper 90-2119, July 1990.

30. Rosenberg, S.D.; and Gage, M.L.: Corrosion Prevention in Hydrocarbon-Fueled Booster Engine Combustion Chamber Liners. AIAA Paper 89-2738, July 1989.

31. Burkhardt, W.M.; Tobin, S.W.; and Mueggenburg, H.H.: Formed Platelet Liner Concept for Regenerative Cooled Chambers. To be published as AIAA Paper 90-2117, July 1990.

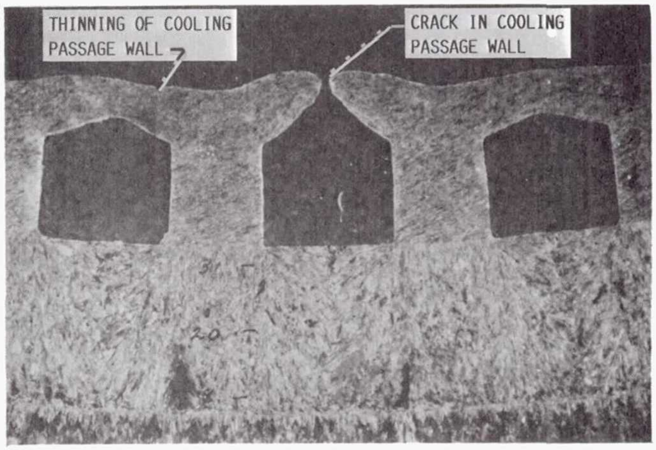


FIGURE 1. - CRACK IN COMBUSTION CHAMBER LINER WALL DUE TO THERMAL RATCHETING. COOLING PASSAGES WERE RECTANGULAR PRIOR TO CYCLIC TESTING.

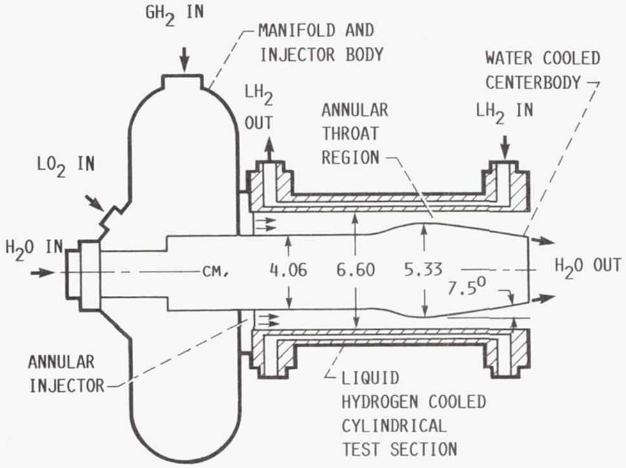


FIGURE 2. - SCHEMATIC OF SUBSCALE ROCKET ENGINE TEST APPARATUS. DIMENSIONS IN CENTIMETERS.

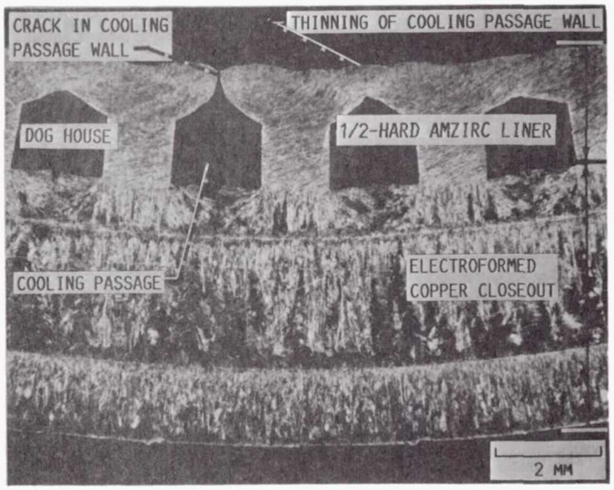


FIGURE 3. - CROSS SECTION OF 1/2-HARD AMZIRC CHAMBER AT THE THROAT PLANE AFTER 393 THERMAL CYCLES. 10x.

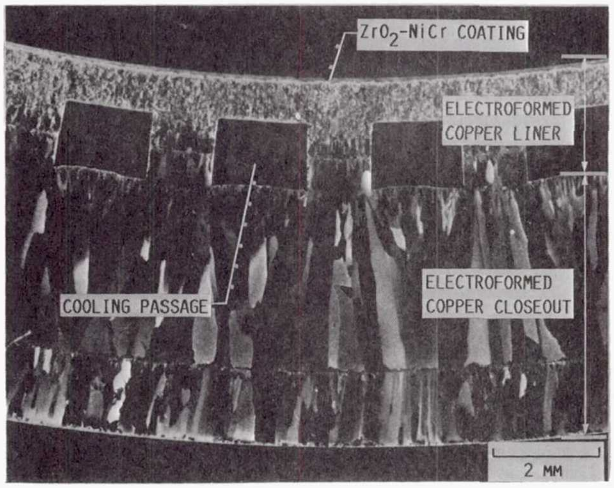


FIGURE 4. - CROSS SECTION OF COATED COPPER CHAMBER AT THE THROAT PLANE AFTER 1450 THERMAL CYCLES. 10x.

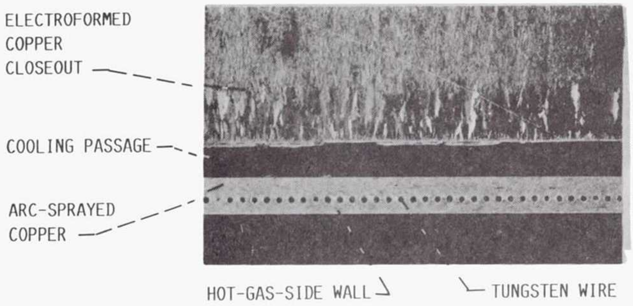


FIGURE 5. - LONGITUDINAL SECTION OF CHAMBER LINER SHOWING TUNGSTEN WIRE REINFORCEMENT.

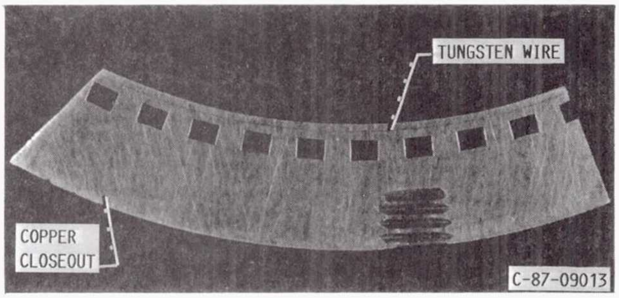


FIGURE 6. - THROAT SECTION OF TUNGSTEN REINFORCED CHAMBER LINER AFTER 400 THERMAL CYCLES.

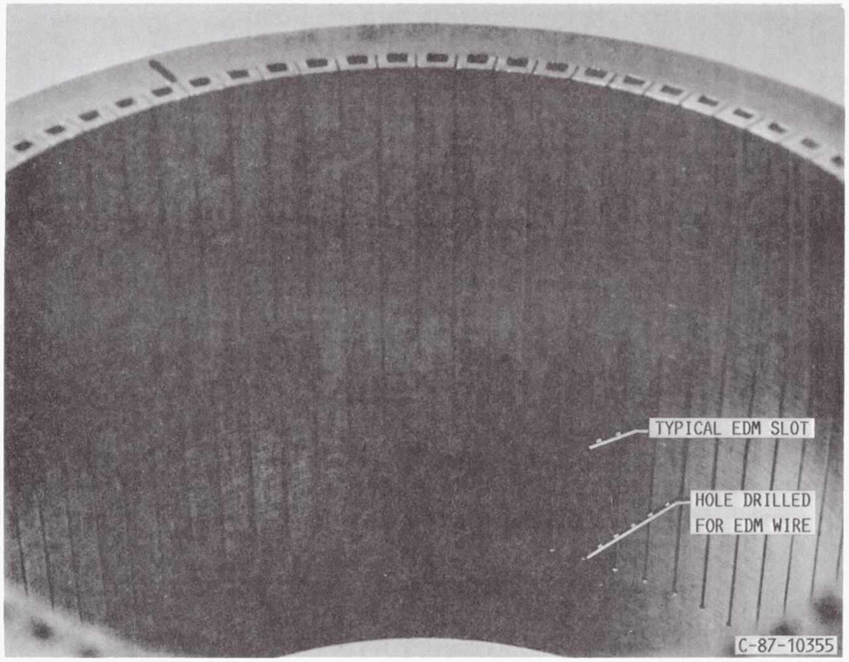


FIGURE 7. - COMBUSTION CHAMBER LINER INCORPORATING ELECTRICAL DISCHARGE MACHINED (EDM) SLOTS.

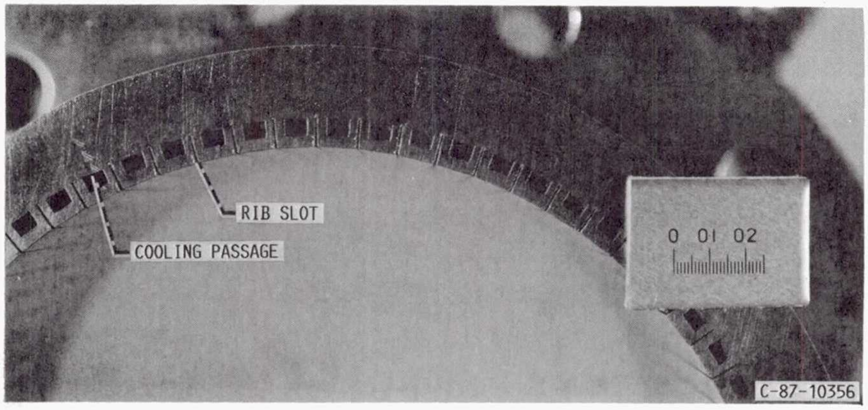


FIGURE 8. - CROSS-SECTION OF CHAMBER LINER SHOWING HOT-GAS SIDE SLOTS.

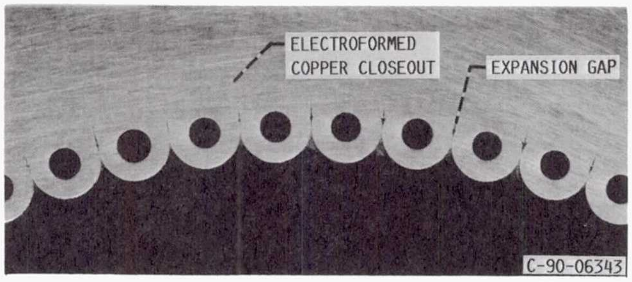


FIGURE 9. - CHAMBER LINER WALL CROSS-SECTION SHOWING ELECTROFORM-FUSED TUBULAR-BUNDLE.

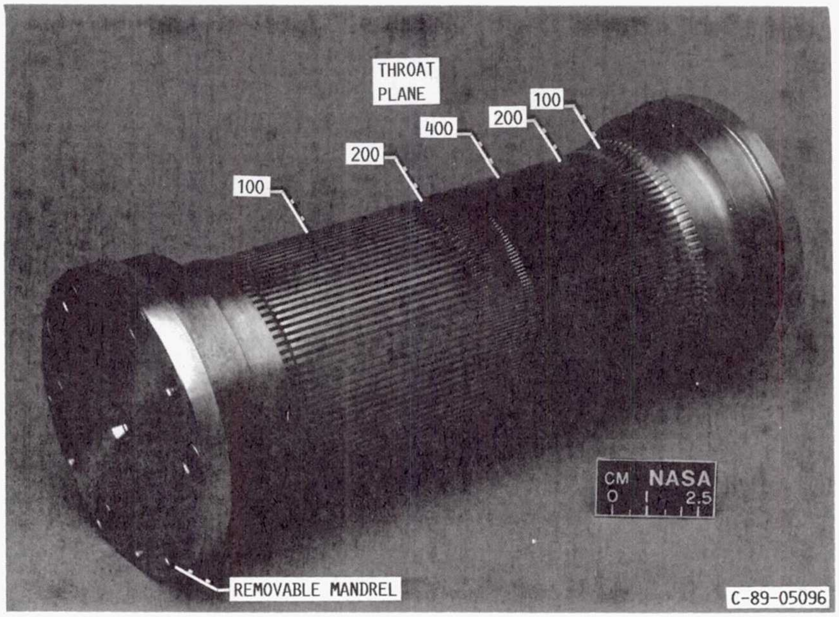


FIGURE 10. - COMBUSTION CHAMBER LINER SHOWING NUMBER OF COOLING PASSAGES AT DIFFERENT LOCATIONS.

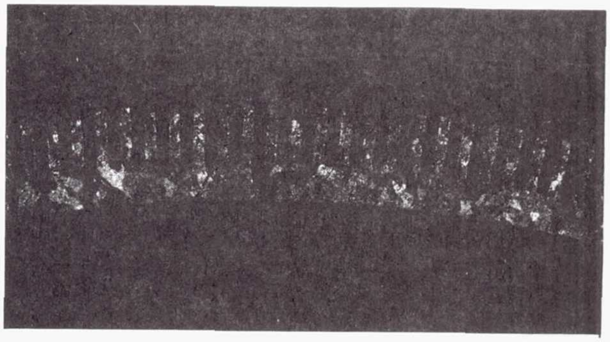


FIGURE 11. - THROAT PLANE OF COMBUSTION CHAMBER LINER SHOWING HIGH-ASPECT-RATIO COOLING CHANNELS.

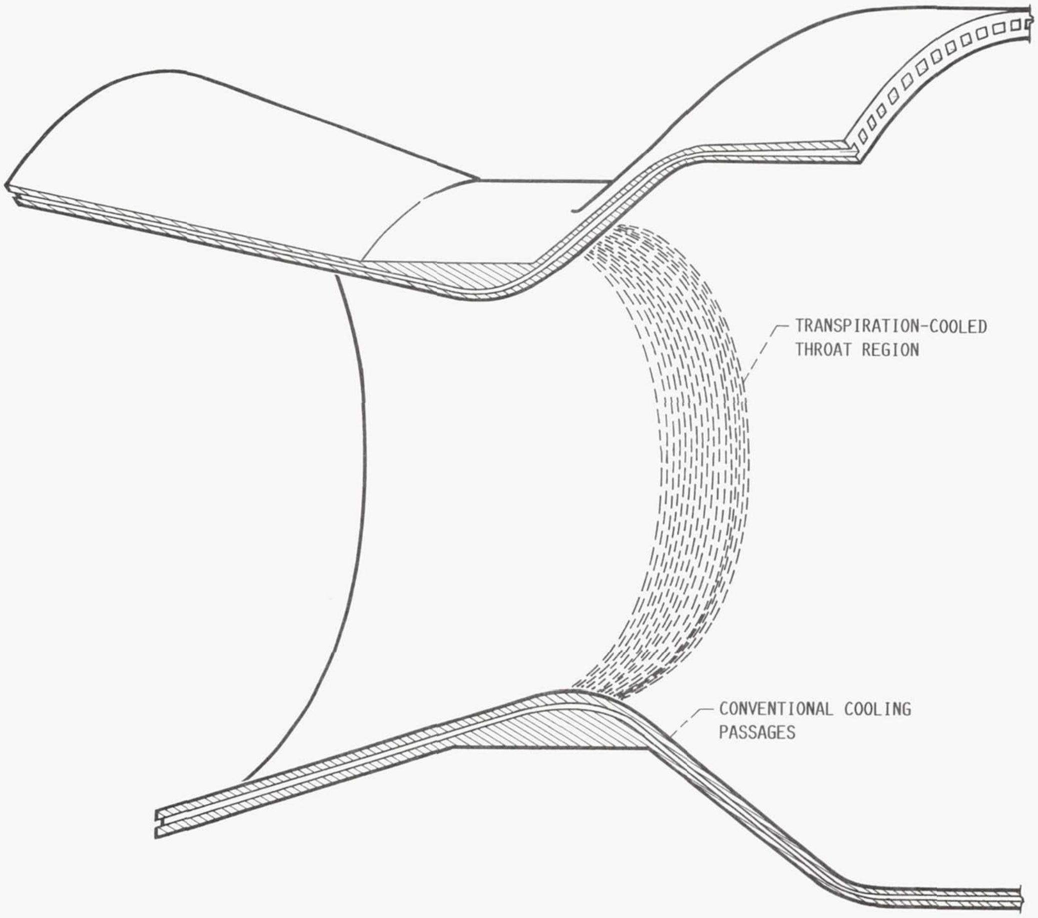


FIGURE 12. - COMBUSTION CHAMBER LINER COMBINING CONVENTIONAL REGENERATIVE COOLING PASSAGES WITH TRANSPIRATION-COOLED THROAT.

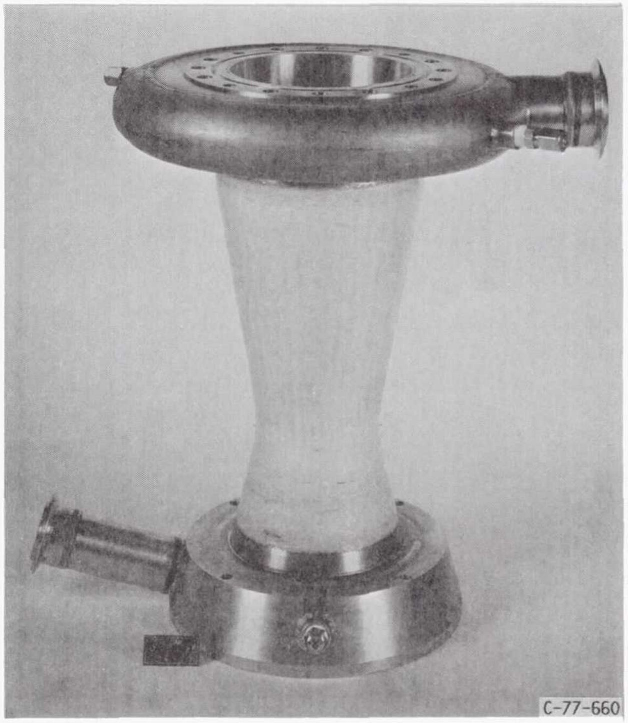


FIGURE 13. - COMBUSTION CHAMBER WITH FIBERGLASS-WRAPPED COPPER CLOSEOUT.

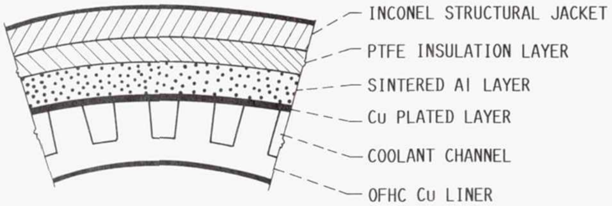


FIGURE 14. - COMBUSTION CHAMBER LINER INCORPORATING A LOW STIFFNESS CLOSEOUT.

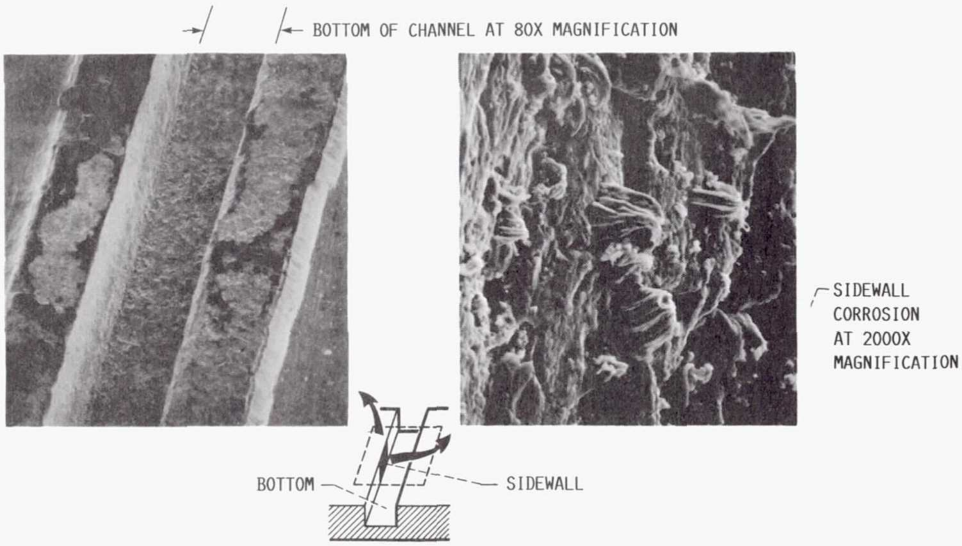
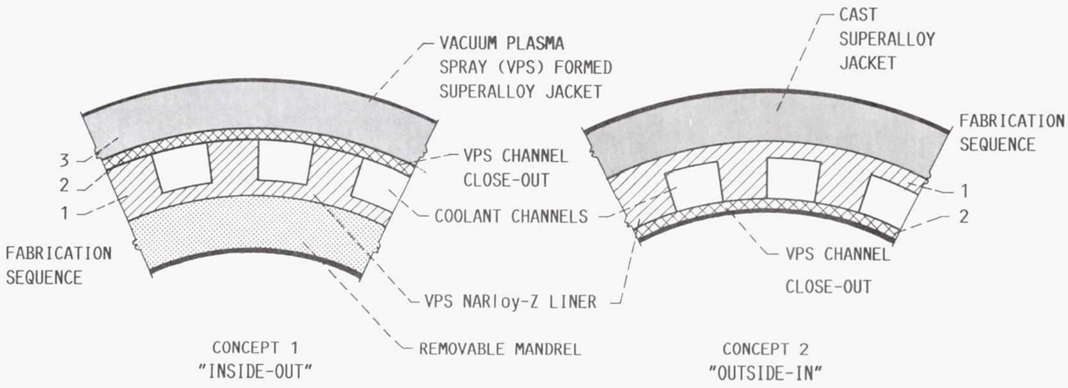


FIGURE 15. - ADDITION OF 50 PPM SULFUR TO RP-1 RESULTED IN CORROSION ON NASA-Zr(Cu, 3.5%Ag, 0.5%Zr) CHANNEL, EVEN AT LOW WALL TEMPERATURE 308 °C (586 °F).



SECTION OF MCC WALL

FIGURE 16. - VACUUM-PLASMA-SPRAYED COMBUSTION CHAMBER LINER FABRICATION.

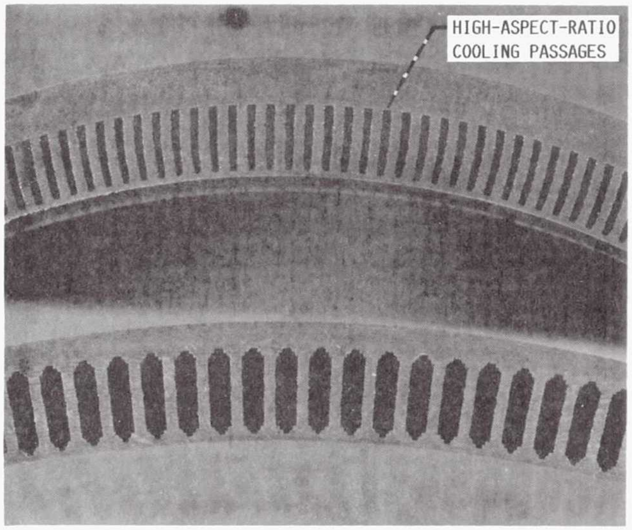


FIGURE 17. - PLATELET-FORMED COMBUSTION CHAMBER LINER SEGMENTS.

NASA
National Aeronautics and
Space Administration

Report Documentation Page

1. Report No. NASA CR-185257 AIAA-90-2116	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Rocket Combustion Chamber Life-Enhancing Design Concepts		5. Report Date July 1990
		6. Performing Organization Code
7. Author(s) Richard J. Quentmeyer		8. Performing Organization Report No. None (E-5564)
		10. Work Unit No. 590-21-11
9. Performing Organization Name and Address Sverdrup Technology, Inc. Lewis Research Center Group 2001 Aerospace Parkway Brook Park, Ohio 44142		11. Contract or Grant No. NAS3-25266
		13. Type of Report and Period Covered Contractor Report Final
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Lewis Research Center Cleveland, Ohio 44135-3191		14. Sponsoring Agency Code

15. Supplementary Notes

Project Manager, Sol H. Gorland, Space Propulsion Technology Division, NASA Lewis Research Center. Prepared for the 26th Joint Propulsion Conference cosponsored by the AIAA, SAE, ASME, and ASEE, Orlando, Florida, July 16-18, 1990.

16. Abstract

NASA continues to pursue technologies which can lead to an increase in life and reduce the costs of fabrication of the Space Shuttle Main Engine. The joint NASA/Air Force Advanced Launch System Program has set its prime objectives to be high reliability and low cost for their new advanced booster engine. In order to meet these objectives, NASA will utilize the results of several ongoing programs to provide the required technologies. This paper presents an overview of those programs which address life-enhancing design concepts for the combustion chamber. Seven different design concepts, which reduce the thermal strain and/or increase the material strength of the combustion-chamber-liner wall are discussed. Subscale rocket test results are presented, where available, for life-enhancing design concepts. Two techniques for reducing chamber fabrication costs are discussed, as well as issues relating to hydrocarbon-fuels/combustion-chamber-liner materials compatibility.

17. Key Words (Suggested by Author(s)) Rocket combustors; Thermal barrier coatings; Composites; Materials compatibility; Vacuum plasma spraying; Chamber liners; Rocket engine cooling	18. Distribution Statement Unclassified – Unlimited Subject Category 20

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of pages	22. Price*
Unclassified	Unclassified	18	A03

NASA FORM 1626 OCT 86

*For sale by the National Technical Information Service, Springfield, Virginia 22161