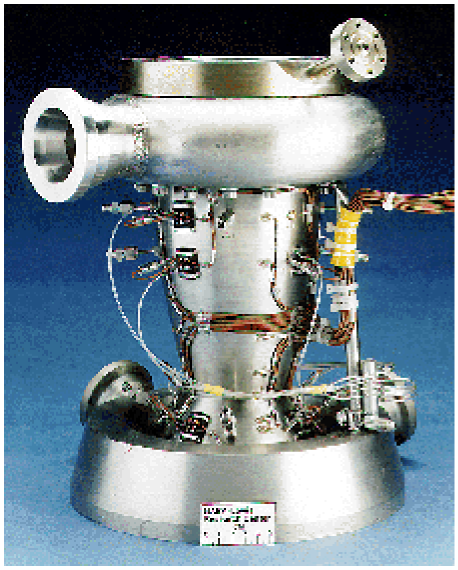
High-Aspect-Ratio Cooling Channel Concept Tested in Lewis' Rocket Engine Test Facility



High-aspect-ratio cooling channel combustion chamber.

Rocket combustion chamber walls are exposed to the high-temperature environment caused by the combustion of propellants. Even with the walls actively cooled by the fuel, the hot gases can deteriorate the walls severely and limit any possibility for reusing the combustion chamber. For many years, the NASA Lewis Research Center has performed subscale investigations of potential improved cooling concepts to extend the life and reliability of the combustion chamber. Results from previous subscale tests have shown that, by increasing the coolant channel height-to-width aspect ratio, the rocket combustion chamber hot-gas-side wall temperature can be reduced by as much as 28 percent, without an increase in the coolant pressure drop (ref. 1). Recently, a series of experiments were completed in Lewis' Rocket Engine Test Facility (RETF) to validate the benefits of high-aspect-ratio cooling channels with a high-pressure, contoured rocket combustion chamber.

Validation of the high-aspect-ratio cooling channel concept was done with a high chamber pressure, contoured combustion chamber (see photo) to simulate the environment of a flight rocket engine. The combustion chamber had 100 conventional coolant channels outside of the critical heat flux area of the combustion chamber throat. These channels had a nominal aspect ratio of 2.5. High-aspect-ratio cooling channels were used in the critical heat flux area. The 100 conventional cooling channels were bifurcated into 200 channels, and their aspect ratios were increased to a range of 5 to 8. The hot-gas-side wall temperature in the throat region was predicted to be approximately 1200 °R. In comparison, a similar combustion chamber with 100 conventional coolant channels throughout the entire combustion chamber length was predicted to have a hot-gas-side wall temperature of approximately 1475 °R. So that the hot-gas-side wall temperature could be verified experimentally, the combustion chamber was heavily instrumented with 28 skin thermocouples, 9 cooling channel rib thermocouples, and 10 cooling channel pressure taps.

The combustion chamber was tested at chamber pressures from 800 to 1600 psia. The propellants were gaseous hydrogen and liquid oxygen at a nominal mixture ratio of 6, and liquid hydrogen was used as the coolant. A total of 29 thermal cycles, each with 1 sec of steady-state combustion, were completed on the combustion chamber. For 25 thermal cycles, the coolant mass flow rate was equal to the fuel mass flow rate. During the remaining four thermal cycles, the coolant mass flow rate was progressively reduced by 6 to 18 percent.

Posttest analysis is being performed on the test data and the combustion chamber. The preliminary results show that the rib thermocouples were cooler than predicted; however, more analysis is required to realize the full effect of this on the hot-gas-side wall temperature. Visual examination of the combustion chamber after testing revealed minimal deterioration in the combustion chamber walls. The final results will be made available so that future rocket engine designers can take advantage of the high-aspect-ratio cooling channel concept.

Bibliography

Carlile, J.A.; and Quentmeyer, R.J.: An Experimental Investigation of High-Aspect-Ratio Cooling Passages. AIAA Paper 92-3154 (NASA TM-105679), 1992.