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Title:
"Performance, Stability and Compatibility of Oxygen/RP-1 Multi-Element Oxidizer-Rich Staged-Combustion Injectors"

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Abstract:
In 2015 and 2016, the National Aeronautics and Space Administration Marshall Space Flight Center designed, fabricated, assembled and hot-fire tested an oxygen/RP-1 propellant multi-element oxidizer-rich staged-combustion test article. The main objective was to provide thrust chamber combustion stability data as part of the Combustion Stability Tool Development program, although demonstration of performance and compatibility of oxidizer-rich main injectors was also important. Funding was provided by the Air Force Space and Missile Systems Center. Five configurations of main injectors were designed and fabricated, using conventional gas-centered swirl coaxial injector element designs generally similar to those used in oxygen/kerosene oxidizer-rich staged combustion engines such as the Russian RD-180 or NK-33 engines. Variations of element features included element size, recess depth, fuel gap width, and the presence of the sleeve separating the swirling fuel flow from the axial oxidizer flow. Ablative combustion chambers were fabricated based on hardware previously used at the NASA MSFC for testing at similar size and pressure. Existing oxygen/RP-1 oxidizer-rich subscale preburner injectors and hot gas ducts from a previous NASA-funded program were modified for use to supply the oxidizer-rich combustion products to the oxidizer circuit of the main injector of the thrust chamber. Testing of the resulting integrated test article – which included the preburner, inter-connecting hot gas duct, main injector, and ablative combustion chamber – was conducted at Test Stand 116 at the East Test Area of the NASA MSFC. The test article was well instrumented with static and dynamic pressure, temperature, and vibration sensors. This paper presents and discusses all the hot-fire test results of the integrated test article thrust chamber. Eighteen successful hot-fire tests of the integrated rig were conducted. Testing was accomplished with all five of the injector element concepts. Main combustion chamber pressures ranged from 710 to 2350 psia, and main combustion chamber mixture ratios ranged from 2.47 to 2.87. A chamber barrier fuel film coolant of about 2% to 4% of the total fuel flow was used for most tests. Characteristic exhaust velocity efficiency excluding the influence of the fuel film cooling ranged from 91% to 98% of theoretical. All tests of the thrust chamber exhibited stable combustion, even down to 40% of nominal operating pressures. Compatibility of the injector face and combustion chamber walls was acceptable. This paper is a follow-on to publication of preliminary test data presented at the 2016 JANNAF Liquid Propulsion Subcommittee meeting.