Evaluation of Vortex Chamber Concepts for Liquid Rocket Engine Applications

Huu P. Trinh
NASA/Marshall Space Flight Center
Huntsville, Alabama

Williams Knuth
Orbital Technologies Corporation
Madison, Wisconsin
And
Scott Michaels
U.S. Army Missile Command
Redstone Arsenal, Alabama

Introduction:

Rocket-based combined-cycle engines (RBBC) being considered at NASA for future-generation launch vehicles feature clusters of small rocket thrusters as part of the engine components[1]. Depending on specific RBBC concepts, these thrusters may be operated at various operating conditions including power level and/or propellant mixture ratio variations. To pursue technology developments for future launch vehicles, NASA/Marshall Space Flight Center (MSFC) is examining vortex chamber concepts for the subject cycle engine application. Past studies indicated that the vortex chamber schemes potentially have a number of advantages over conventional chamber methods. Due to the nature of the vortex flow, relatively cooler propellant streams tend to flow along the chamber wall. Hence, the thruster chamber can be operated without the need of any cooling techniques. This vortex flow also creates strong turbulence, which promotes the propellant mixing process. Consequently, the subject chamber concepts not only offer the system simplicity but they also would enhance the combustion performance. The test results showed that the chamber performance was markedly high even at a low chamber length-to- diameter ratio (L/D) [2]. This incentive can be translated to a convenience in the thrust chamber packaging.

Vortex chamber concepts for bi-liquid propellant systems were first introduced in 1960[2]. In this considered system, one of the propellants is injected tangentially to the inner chamber diameter. That develops a rotational flow pattern along the chamber wall. The second propellant is introduced either tangentially at the periphery or radially at the center of the chamber. This injecting scheme creates vortex flows, which serve the function of breaking up, vaporizing, and mixing propellants. Hot-fire tests for a small vortex thruster were conducted. The results showed that excellent combustion efficiencies, as high as 98%, were obtained. The wall heat transfer flux with the vortex injectors was significantly lower than with their conventional counterpart impinging injectors. These investigations focused on missile applications, in which hypergolic propellant systems were employed. These efforts were then extended to demonstrate a throttleability feature in conjunction with the utilization of this concept. A 10:1 throttling was accomplished by varying the injection areas of both propellants simultaneously. An average of 94% in combustion efficiency was recorded over the throttling range[3]. Later, a

more rigorous study on the throttleable vortex engine concept was performed with a variety of hypergolic liquid propellant systems[4].

Deep throttling (50 to 1) with the low L/D vortex chamber was demonstrated by Wilson in 1970[5]. Wilson investigated various injection schemes for hypergolic liquid as well as gel propellant systems. He selected a tangential-tangential injection arrangement for his study since this injector configuration was simple in fabrication and it was able to deliver a specific impulse (Isp), which exceeded 90% of the theoretical predicted value. In this injection technique, both propellants were injected along the chamber wall and tangential to the inner wall diameter. Both propellant orifices were on the same chamber cross-section plane. Wilson was able to achieve a wide range of throttling using a displacement venturi system. He also observed that the torque force created by the tangential injection was significantly smaller than the axial force.

A recent investigation of the low L/D vortex chamber concept for gel propellants was conducted by Michaels[6]. He used both triplet (two oxidizer and one fuel orifices) and unlike impinging schemes to inject propellants tangentially along the chamber wall. The test results showed that an excellent performance, with an Isp efficiency of 92%, can be achieved.

Although most of previously discussed studies suggested that the vortex chambers provided relatively cool flows in the near wall region, the hot-fire tests were conducted either with water-cooled chambers or at short testing durations. Ablative materials were sometimes used for critical thruster components, such as the faceplate at the head end, throat, and nozzle sections. To achieve the advantage of the cold wall feature of the vortex chambers, innovative vortex injection schemes were examined at Orbital Technical Corporation[7]. By injecting the oxidizer tangentially at the chamber convergence and fuel axially at the chamber head end, Knuth et al. were able to keep the wall being relatively cold. This concept was employed in tests at which gaseous oxygen and hydrogen at near-stoichiometric mixture ratios were combusted in plexiglas chambers. The post inspection showed no appearance of overheating on the plexiglas chambers.

This paper reports the progress of the efforts at MSFC to examine the latter two injection schemes. Thrust chamber performance and thermal wall compatibility will be evaluated. The chamber pressures, wall temperatures, thrust, and exhaust plume flow fields will be measured as appropriate. The test data will be used to validate CFD models, which, in turn, will be used to design the optimum vortex chambers. Although the primary application of the vortex chamber concept is for the RBBC engines, the success of this technology will lead to these concepts being applied to other future liquid rocket engines.

The Vortex Combustion Cold-Wall (VCCW) Chamber:

- **Concept Description:**

This vortex combustion chamber generates a co-axial rotational flow field in which a propellant is caused to swirl along the chamber wall and upward to the chamber head end

by injecting the fluid tangent to the wall near the chamber convergence. A typical flow at this location, as shown in figure 1, has been simulated with a CFD code. Upon reaching the chamber head end, the fluid migrates inwards to form a separate vortex spiraling downward. Fuel fluid is then injected into the core vortex to be mixed with the oxidizer. The combustion products are retained in the core vortex and do not contact the chamber wall until near the exit nozzle. In this case the chamber walls are cooled by the oxidizer, hence the term cold wall chamber.

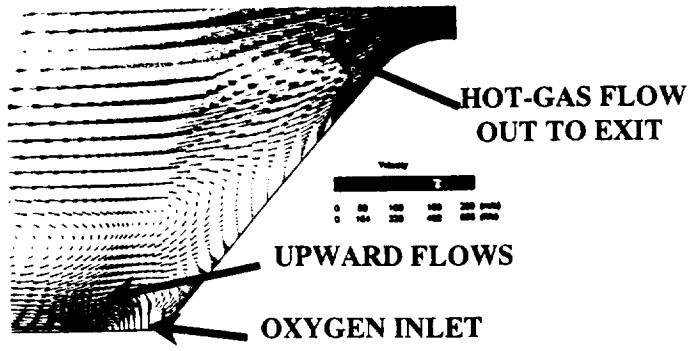

Figure 1. CFD Simulation of the Vortex Flow Field Near the Chamber Throat.

• **Recent Test Activities:**
The oxidizer injected tangentially to the chamber wall at the chamber convergence was first introduced in studies of hybrid rocket engines in which the wall served as the fuel[8, 9,10]. This injection technique was then used to develop the VCCW chamber concept. Cold flow tests using water and air were conducted to characterize the internal flow structures. To verify the cold wall feature, hot-fire tests, as shown in figure 2, have also been performed on a plexiglas chamber with the fuel injected only into the core vortex and the oxidizer injected tangentially to the chamber wall. Cold wall chamber testing included oxygen/hydrogen, and oxygen/kerosene in the rocket mode, and air/kerosene and air/hydrogen in the ramjet mode[7,11].

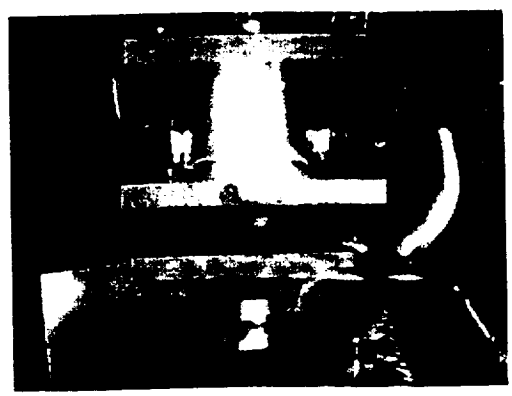

Figure 2. Gaseous Oxygen/Gaseous Hydrogen Cold Wall chamber Firing.

Key research findings to date include:

- The cold flow testing showed the presence of two vortices, an inner vortex and an outer vortex, both of these vortices rotate in the same direction. The visualization data also suggested that there is very little if any cross flow other than at the head end of the chamber.
- Cold wall testing has been achieved with oxygen/hydrogen in plexiglas chambers at near-stoichiometric mixture ratios, with chamber pressures in excess of 100 psig. These tests illustrate the effectiveness of the flow field in protecting the chamber wall from the intense combustion temperatures in the core flame.

- Ramjet testing has also been successful, and has demonstrated the ability of a simple flame holder to anchor the flame at the head end of the chamber[11].

The Impinging Stream Vortex Chamber:

- **Concept Description:**

The impinging stream vortex engine (ISVE) offers an alternative to increasing the length and volume of conventional combustion chambers. The ISVE is radically different from the conventional impinging stream liquid/gel engine in that the propellants are injected tangentially to the chamber wall, impinged, and then swirled via the vortex flow that is established with the tangential injection. The initial mixing occurs during stream impingement and the final mixing takes place in the highly turbulent vortex region between the injector orifices and the nozzle throat. There has been some evidence from post-test examination of the engine hardware tested with the gel propellants that the heavier solid particles separate from the gas particles and move toward the chamber wall. It has been postulated that centrifugal movement of the solid particles increases the path length and thus increases the time in the combustion zone. An additional attribute of this injection concept is that the propellants provide transpiration cooling at the injector area and protect the cylindrical chamber wall from the combustion flame.

- **Recent Test Activities:**

Since the ISVE is a relatively new concept, the databases and analytical models relating performance to the engine design parameters are essentially nonexistent. To address design issues, the U.S. Army Aviation and Missile Command (AMCOM) has recently conducted limited testing of the vortex engine. The tests, as shown in figure 3, have yielded delivered specific impulse efficiencies in excess of 95 percent of theoretical using inhibited red fuming nitric acid (IRFNA) as the oxidizer and 50 percent carbon loaded monomethylhydrazine (MMH) as the fuel.

Figure 3: Typical Hot-fire test of the ISVE with gel propellants

Currently, AMCOM initiates an experimental effort, which focuses on static testing eight vortex engines of various configurations. The resulting two-level fractional factorial test matrix consists of eight unique engines that are configured with various propellant injection angles, fuel gel particulate loadings, nozzle entrance angles, chamber volume to the throat area ratios (L^*), chamber length-to-diameter ratios (L/D), and numbers of unlike doublet orifice pairs. The eight engines were designed, fabricated, and are currently being static tested. The static test data, as it becomes available, will be used to validate CFD models that can be used to virtually determine optimum engine configurations proposed for future tactical systems.

Operating condition baseline for tests:

Several RBBC engine schemes are being considered at NASA. Depending on the individual RBBC engine scheme, the operating conditions and thrust level may be varied. In addition, various propellant systems are being evaluated for the subject engine cycle. In 1999, MSFC conducted an in-house study of a hydrocarbon-fueled axisymmetric ejector ramjet engine for the purpose of the RBBC technology demonstration[12]. The engine project was named DRACO (Demonstration of Rocket and Airbreathing Combined-cycle Operation). Although the operating condition baseline for testing the vortex chamber concepts, as shown in table 1, is primarily derived from the subject study, this baseline is similar to the other ones on considered RBBC schemes. Liquid oxygen (LOX) and hydrocarbon fuel (RP-1) propellants were selected for this chamber technology demonstration. However, this concept can be applied to other propellant systems.

Table 1: Baseline Operating Conditions

PARAMETERS	UNIT	VALUES
Chamber Pressure	psia	1000
Ideal Vacuum Thrust	lbf	1250
Mixture Ratio		2.6
LOX	lb/sec	3
RP-1	lb/sec	1.15
Throat diameter ~	in	0.96 - 1.00
Chamber Diameter	in	3
Nozzle Exit Diameter ~	in	3.11 - 3.50

Test and Measurements:

The VCCW chamber will be tested at Orbital Technologies Incorporation (Orbitec). Orbitec is testing several injection configurations and arrangements on a plexiglas chamber using gaseous oxygen and kerosene. The hot-fire tests have been operated with low chamber pressures in the order of 100 psia. The vortex chamber flow structures and the flame holder locations have been characterized from the pressure and temperature measurements and visualization data. Orbitec will aggressively perform hot-fire test series on other test hardware to cover the previously described operating conditions. Because of the facility limitation, Orbitec plans to test the vortex chamber up to 10 seconds at the main stage.

MSFC and the U. S. Army are jointly investigating an application of the ISVE concept for the cryogenic oxygen/hydrocarbon propellant system. This vortex chamber concept is currently tested with gel propellants at AMCOM at Redstone Arsenal, Alabama. A version of this concept for the LOX/RP-1 system, as shown in figure 4, has been derived from the one for the gel propellant.

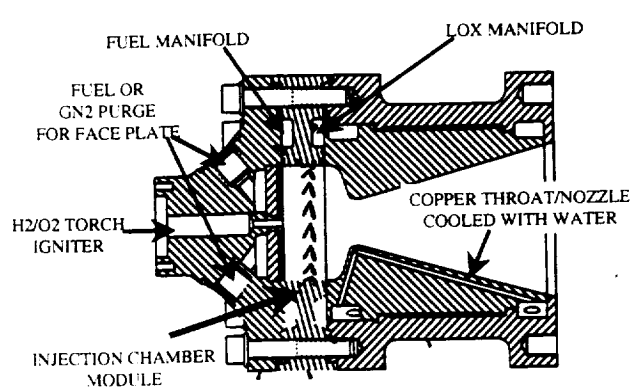

Figure 4: Design of Vortex Chamber for LOX/RP-1

An unlike impinging injector was employed to deliver the propellants to the chamber. MSFC is also conducting alternative injection schemes associated with the U.S. vortex chamber concept. Long duration tests (approximately up to 50 seconds) will be conducted on the ISVE to study the thermal effects.

Measurements in the previous tests showed that the chamber pressures vary drastically along the chamber wall. This is due to the existence of the vortices in the chamber flow field. Hence, the combustion efficiency may not be easily determined. For these two projects, measured thrust data will be collected. The performance comparison will be in terms of specific impulse efficiencies. Orbitec will also install thermocouples and pressure transducers along the chamber wall to collect thermal and flow conditions.

In addition to the thrust measurements on the U.S. Army vortex chamber, several pressure and temperature readings at various locations on the faceplate chamber head will be made. Due to the short chamber length and the propellant manifolds located along the outside of the chamber wall, intrusive measurements can't be used on the chamber wall. Instead the exhaust plume flow field will be measured using either the Raman scattering or the emission/absorption techniques. Spatial distributions of combustion products and flow field temperature on the exhaust plume will be recorded.

Summary and Discussion:

- **Potential Benefit:**

Success of the exploitation of the vortex chamber concepts undoubtedly will have huge benefits for future rocket engine designs. Due to the centrifugal force, the axial swirling flows created by the tangential injection schemes tend to keep a high-density fluid, and therefore, relatively cooler flows, in the near chamber wall region. Hence, this injection approach does not require any traditional cooling methods, such as cooling wall channels wall and/or film cooling techniques. Consequently, turbo-machinery systems do not need to provide any extra power to overcome the pressure loss associated with propellant flowing through the cooling channels. This advantage alone would significantly simplify the thrust chamber system and, at the same time, also increase the engine life. No film cooling needed in the combustion chamber translates to no combustion performance loss due to the mixture ratio striation effect. Moreover, the vortices generate highly turbulent flows and long flow paths, which, in turn, enhance the propellant mixing and combustion processes. For the ISVE concept, a high thrust performance can also be achieved even at a low L/D chamber configuration. This will offer an advantage for the engine packaging.

The successful demonstration of applying the vortex injector on a small-scaled hybrid rocket engine has shown that uniform solid fuel burning as well as a high solid-fuel regression burning rate can be achieved[8,9,10]. Orbitec has also tested the VCCW chambers with various gaseous and gaseous/liquid propellant combinations[7,11]. On the other hand, the U.S. Army is actively exploring the ISVE concepts for missile rocket engine applications, for which a number of propellant systems including hypergolic liquid and gel propellants have been considered. All these efforts have indicated that the vortex chamber technology potentially has broad applications for various rocket engine types.

• **Future Activity:**

Currently, Orbitec is studying in detail, several injection techniques at low chamber pressure conditions using gaseous oxygen and liquid kerosene. This is due to the present facility limitation at Orbitec. The test facility is being upgraded for the cryogenic oxygen and RP-1 system. The first test series will start in December of 2000. The test objective is to demonstrate the feasibility of the VCCW chamber for the liquid oxygen/RP-1 system. In the present phase of this project, limited effort will be dedicated for characterizing the effect of the injector/chamber configuration on the chamber performance and the wall thermal compatibility.

For the ISVE concept, a first injector and chamber hardware were designed based on the available data and experience of the gel propellant systems, which were undertaken at the U.S. Army. The hardware fabrication is in progress. Alternate injection schemes for the ISVE are being analyzed. Hot-fire tests of the vortex chamber will start in February of 2001 at the TS115 facility of MSFC. The ISVE tests have a similar objective as described for the VCCW chamber test activity. The data from both testing efforts will be used to verify the CFD models as well as to derive optimum designs in the future.

References:

1. Daines, R. and Segal, C., Combined Rocket and Airbreathing Propulsion Systems for Space-Launch Applications, Journal of Propulsion and Power, Vol. 14, No. 5, Sept.-Oct. 1998, pp. 605-612.
2. Dawson, B., Feasibility Study on Vortex Combustion, Report RMD 1165-Q5, Contract DA-30-069-ORD-2772, Report Period: June 25 – Sept. 24, 1960.
3. Feigel H. and Fabbro A., Controllable Thrust Chamber Study Program, Task III, Varitex Prototype, Final Report RMD 6020-F, Contract NAS 7-114, Jan. 1963.
4. Chatfield, D.S., Radially Distributed Annular Combustor For Packaged Liquid Rocket Engines (U), (Radial State of the Art Program), Final Report RMD 6048-F, Contract Now 65-0540-c, April 1967.
5. Wilson, B., All-Fluid Throttling of The Low L/D Vortex Engine, 12th JANNAF Liquid Propulsion Meeting, CPIA Pub. 201, Vol. 1, Oct., 1970, pp. 69-85.
6. Michaels, R. and Wilson, B., The Low L/D Vortex Engine for Gel Propellants, The 1995 JANNAF Gel Propulsion Technology Symposium, Sept. 19-21, 1995.
7. Knuth W., et al., Vortex Combustion O2/HC Rocket Engine Development, Phase III SBIR Contract with NASA, NAS8-00116.
8. Knuth, W. J., et al., Preliminary CFD Analysis of The Vortex Hybrid Rocket Chamber and Nozzle Flow Field, AIAA98-3351, 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cleveland, OH, July 13-15, 1998.
9. Knuth, W. J., et al., Development and Testing of a Vortex-Driven, High-Regression Rate Hybrid Rocket Engine, AIAA98-3507, 34th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cleveland, OH, July 13-15, 1998.
10. Knuth, W. J., et al., Experimental Investigation of a Vortex-Driven, High-Regression Rate Hybrid Rocket Engine, AIAA98-3348, 34th

AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Cleveland, OH, July 13-15, 1998.

11. Knuth, W., et al., "Vortex Combustion Ramjet", NASA Contract NAS3-99039, OTC-G075-FR-1, Orbital Technologies Corporation, Madison, WI, June 15, 1999.
12. Turner J., McArthur C., DRACO: Development Activities on a Hydrocarbon-Fueled Ejector Ramjet RBBC Engine, 1999 JANNAF/CS/APS/PSHS Joint Meeting October 18-21, 1999.