Chapter 10:
Large-Scale Hybrid Motor Testing

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Introduction

Hybrid rocket motors can be successfully demonstrated at a small scale virtually anywhere. There have been many suitcase sized portable test stands assembled for demonstration of hybrids. They show the safety of hybrid rockets to the audiences. These small show motors and small laboratory scale motors can give comparative burn rate data for development of different fuel/oxidizer combinations, however questions that are always asked when hybrids are mentioned for large scale applications are – how do they scale and has it been shown in a large motor? To answer those questions, large scale motor testing is required to verify the hybrid motor at its true size.

The necessity to conduct large-scale hybrid rocket motor tests to validate the burn rate from the small motors to application size has been documented in several places[1,2,3]. Comparison of small scale hybrid data to that of larger scale data indicates that the fuel burn rate goes down with increasing port size, even with the same oxidizer flux. This trend holds for conventional hybrid motors with forward oxidizer injection and HTPB based fuels. While the reason this is occurring would make a great paper or study or thesis, it is not thoroughly understood at this time. Potential causes include the fact that since hybrid combustion is boundary layer driven, the larger port sizes reduce the interaction (radiation, mixing and heat transfer) from the core region of the port.

This chapter focuses on some of the large, prototype sized testing of hybrid motors. The largest motors tested have been AMROC's 250K-lb_f thrust motor at Edwards Air Force Base and the Hybrid Propulsion Demonstration Program's 250K-lb_f thrust motor at Stennis Space Center. Numerous smaller tests were performed to support the burn rate, stability and scaling concepts that went into the development of those large motors.

Nomenclature/Acronyms

AMROC: American Rocket Company

Btu: British Thermal Unit

CP: Center Port

C*: Characteristic exhaust velocity

DARPA: Defense Advanced Research Projects Agency

FLOX: Fluorine and Liquid Oxygen mixture

GOX: Gaseous oxygen

HPDP: Hybrid Propulsion Demonstration Program

HTPB: Hydroxyl Terminated Polybutadiene

ISP: Specific impulse

JIRAD: Joint Industry Research and Development

LOX: Liquid oxygen

MSFC: Marshall Space Flight Center

NAI: non-acoustic instability

O/F: Oxidizer to fuel ratio

PSD: Power Spectral Density

SET-1: Single Engine Test vehicle developed by AMROC

SSC: Stennis Space Center

TEA/TEB: Triethylaluminum/Triethylborane

TEAL: Triethylaluminum

TNT: Trinitrotoluene

UTC: United Technologies Corporation

Back ground - Why hybrids?

Hybrids, considered part solid and part liquid propulsion system, have been caught in the middle of development goals of the various NASA and military programs. Solid rocket motor technology has matured due to the advantages of design simplicity, on-demand operational characteristics and moderately low cost. The reliability of solids, given minimal maintenance requirements, made them the ideal system for military applications. On the other hand, liquid rocket engine technology has matured due to their higher specific impulse (ISP) over solids and variable control thrust capability.

Prior to 2004, hybrid rockets have been used in only one flight-production application (Teledyne Ryan AQM-81A Firebolt Supersonic Aerial Target). Their recent successful application to a manned flight demonstration (Burt Rutan's SpaceshipOne), may suggest that advantages have been overlooked in some potential applications, and hybrids may be getting renewed interest. Hybrid rockets inherently combine the safety features of a liquid propulsion system (throttle, shut-down, restart) while deriving the cost and operational benefits of a solid propulsion system. Specific details regarding these advantages include the following:

Handling – Virtually all hybrids fuels are considered inert (Class 1.4c propellant – zero TNT equivalent), that is they can be transported via normal shipping techniques with no additional safety requirements. This is a significant benefit when compared to traditional solids, where any processing is considered a hazardous operation and special handling considerations must be observed.

Casting – Classical hybrid motors can be cast in light industrial facilities employing the techniques used in traditional solid propellant casting. Hybrids are largely insensitive to cracks and defects in the propellant, but gross disturbances in the flow from air bubbles cast in the fuel (voids) can cause problems during hot-fire operations.

Simplicity – Hybrid rockets are more complex than solids due to the need for an oxidizer delivery system, with an associated oxidizer tank pressurization system and pump if necessary. Although hybrids are more complex than solids, they use only one fluid system, which make them less complex than liquid bi-propellant systems (liquid rocket engines).

Throttling – Hybrids can be throttled by increasing the oxidizer flow rate via varying the opening of the oxidizer valve in a pressure fed system or speeding the pump in a pump fed system. Since the fuel regression rate is a function of the oxidizer flux, lowering the oxidizer flow rate lowers the fuel regression rate and resultant thrust level.

Restart – Hybrid motors can typically be ignited many times, until the fuel grain is consumed or the nozzle and other components are past their design life limits.

Performance – The ISP of a Hydroxyl Terminated Polybutadiene - LOX rocket is equivalent to a RP-1-LOX engine, and significantly higher than a solid rocket motor.[4] Other fuel and oxidizer combinations yield higher and lower performance values, with different system issues to work with.[5] To a certain extent, performance can be tailored utilizing fuel additives or other propellant modifications to meet specific requirements.

Cost – The handling and casting process costs should be significantly lower than that of a solid, with no oxidizer in the fuel and therefore lower safety concerns. Since there is only one liquid propellant used, the system costs should be significantly less than that of a liquid system.

High Energy Hybrid Space Engines

Space propulsion systems typically use solid rockets, with relatively low ISP but high density impulse or LOX-hydrogen systems with high ISP but lower density impulse. In the 1960s, NASA decided to investigate hybrid rockets to see if a high ISP and high density system could be developed.[6]

NASA selected UTC (United Technologies Corporation) to perform a series of investigations devoted to high-energy space engines. One concept was based on the utilization of the very energetic reaction between lithium and fluorine, two elements at the opposite ends of the Mendeleev Periodic table. The lithium was incorporated into an HTPB binder and the fluorine was mixed with oxygen to create FLOX, optimizing the performance of the system.[6]

In order to satisfy conductivity restrictions of the binder and avoid melting of the lithium, an eventual composition incorporating a combination of lithium and lithium hydride was finally developed for the propulsion system shown in Figure 1. This throttleable system burned smoothly and exhibited very high performance. Its ground performance converted to vacuum ISP, with an area ratio of 40:1, would be in excess of 400 seconds. A picture of this firing is shown in Figure 2 which appeared on the cover of the January 26, 1970 issue of Aviation week.[6]

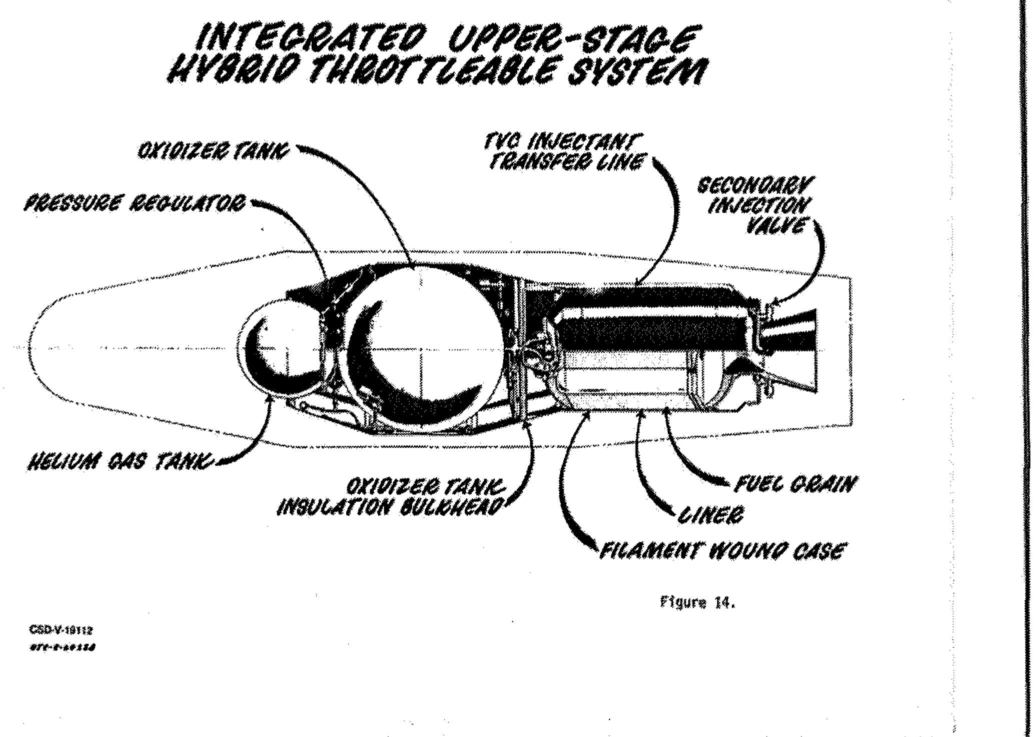


Figure 1 UTC Hybrid Upperstage Concept[6]

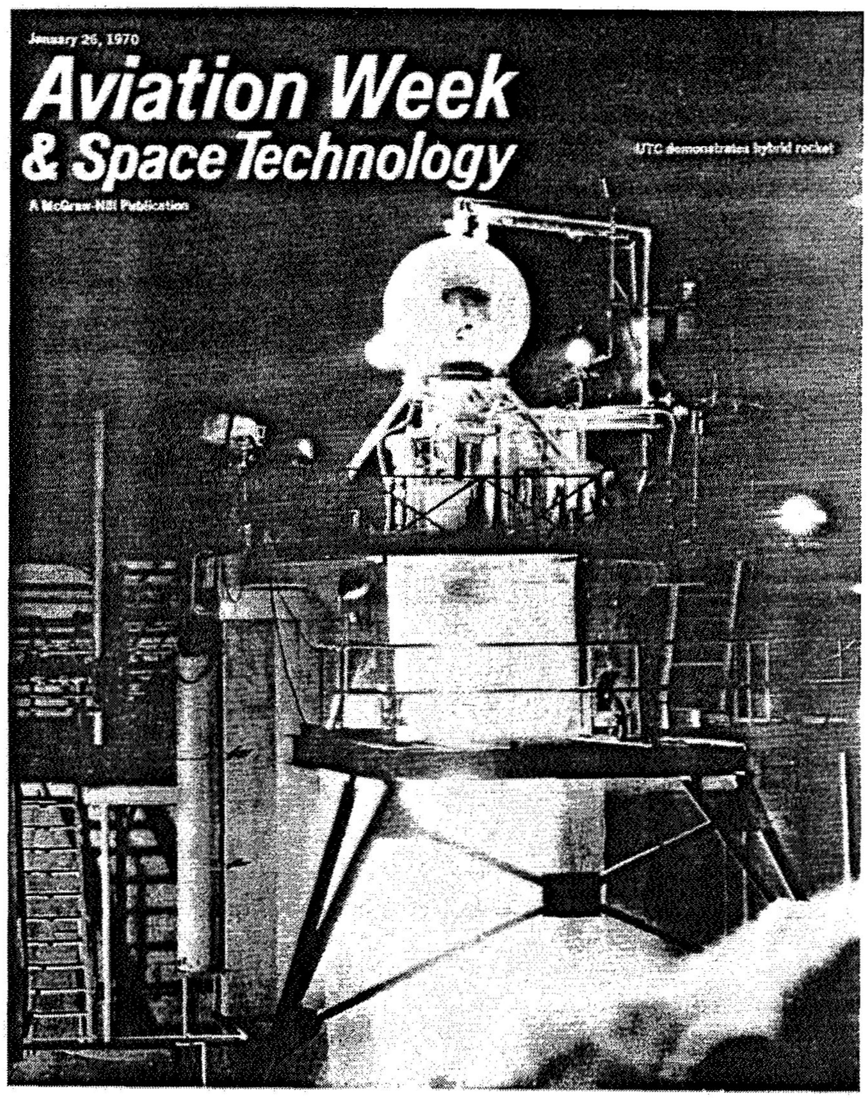


Figure 2 UTC Hybrid Firing[6]

AMROC Experience

AMROC, the American Rocket Company, was for years the leader in hybrid rocket development. Their dream was a low-cost commercial vehicle for entry into space. During the time AMROC was financially solvent, they did exciting work on a large range of sizes of hybrid rocket motors. They published numerous hybrid papers to generate interest in hybrids; however they still kept their trade secrets close. The AMROC intellectual property rights and data were eventually sold to Space Dev, who are currently pursuing hybrids.

AMROC Combustion Stability at 75,000K-lb$_f$ thrust

Initial development activity at AMROC targeted a 75K-lb$_f$ thrust motor for suborbital payload delivery on the Single Engine Test (SET) launch vehicle. During the development of the 75,000 pound thrust H-500 motor, AMROC used three sizes of test motors:[7]

1) Small Scale – The small scale motors were ~18"-diameter motors producing 10,000 pounds thrust. These small scale motors were used for fuel formulation, insulation, and other materials for compatibility testing. 19 of these motors were built and fired.

2) Half Scale – The half scale motors were ~36"-diameter motors producing 33,000 pounds thrust. These motors were used to test different concepts and issues in multi-port grain development. 3 of these motors were built and fired.

3) Full scale – The full scale motors were ~51"-diameter motors producing 75,000 pounds of thrust. These were tested in heavy-weight cases and in flight-weight composite cases. Twelve of these motors were built; one a manufacturing pathfinder and the remainder were fired. Details of the tests run in each configuration is shown in Table 1.[7]

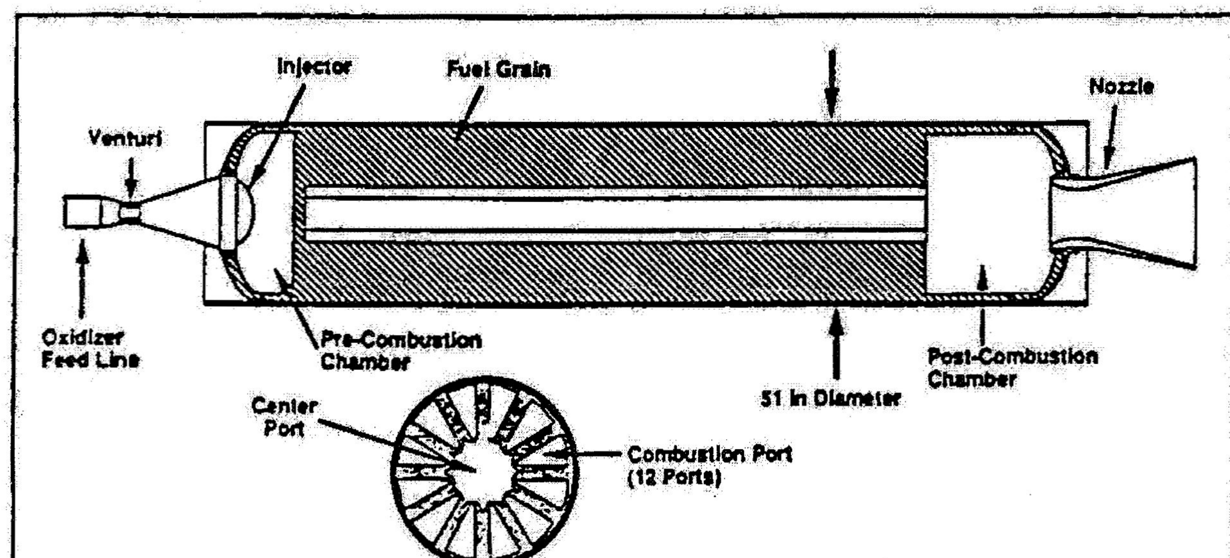


Figure 1. AMROC Hybrid Rocket Motor

Figure 3 AMROC 75K-lb$_f$ Hybrid Motor[7]

Table 1AMROC 75K-lbf Motor Development Testing[7]

Motor	First Firing	No. of Firings	Comments
H-001	4 Dec 86	4	33K-lb$_f$ motor
H-002	17 Feb 87	3	33K-lb$_f$ motor
H-003	21 Apr 87	4	Last 33K-lb$_f$ motor
H-004	14 Oct 87	5	First 75K-lb$_f$ motor
D-001	18 Apr 88	6	Fiberglass motor case
H-005	23 Dec 87	5	
H-006		0	Manufacturing Pathfinder. Not Fired.
H-007	28 Jun 88	3	
H-008	26 Jul 88	5	
H-009	2 Sep 88	1	First full-duration firing.
H-010	25 Oct 88	6	

H-011	19 Dec 88	1	
H-012	13 Mar 89	1	
H-013	18 Apr 89	1	
DF-1	11 Jul 89	3	Qualification test Motor
DF-2	5 Oct 89	1	Set-1 flight motor

AMROC found non-acoustic instability (NAI) during the development of a large-scale hybrid rocket motor. The frequencies of the oscillations were related to the fill/flush time of the combustion chamber. The observed frequencies were similar to, though slightly higher than, those predicted by solid rocket motor correlations. The magnitude of the oscillation was governed by the oxidizer feed system and injector characteristics.[7]

Stability problems were first encountered by AMROC during the 33K-lb_f motor firings. Since their 10K-lb_f motors were relatively immune to instabilities, AMROC reasoned that the physical mechanisms driving the oscillations were enhanced at larger scale. Therefore, solving the stability problem at half-scale would not guarantee that the full-scale motor to be free of oscillation. AMROC decided to work on the stability problems using the larger 75K-lb_f motor. A series of tests were performed using the 75K-lb_f motor to identify the cause of and eliminate the combustion oscillations.[7]

AMROC identified Inadequate LOX vaporization as the major cause of NAI. This points out the need for either a precombustion (vaporization) chamber upstream of the combustion ports to allow for adequate gasification of the LOX or injecting the oxidizer in gaseous form. For liquid injection, reduced droplet size was more important than low axial velocity in increasing droplet vaporization. Splashblocks, defined as sacrificial fuel surface areas downstream of the LOX injector, were used to increase the residence time in the vaporization chamber and found to be effective in suppressing NAI. AMROC found that the required area of forward facing surface was 13% of the total motor cross sectional area.[7]

AMROC demonstrated their theories on the qualification test motor (DF-01), which employed the enhanced techniques to suppress non-acoustic instabilities. Changes from early full scale motors included:

-Increased venturi pressure drop to increase feed system capacitance,
-Decreased injector manifold volume to reduce injector capacitance,
-Increased injector pressure drop to reduce droplet size,
-Use of splashblocks to increase effective droplet residence time.

With these modifications, the qualification test motors met all requirements for combustion stability.[7]

AMROC 250K-lb_f Motor Development

After the failed launch of SET-1 on October 5, 1989, which was built around the 75,000-lb_f hybrid motor, AMROC reevaluated the market and started to design a larger, 250,000-lb_f hybrid rocket motor for a different sized launch vehicle.[8]

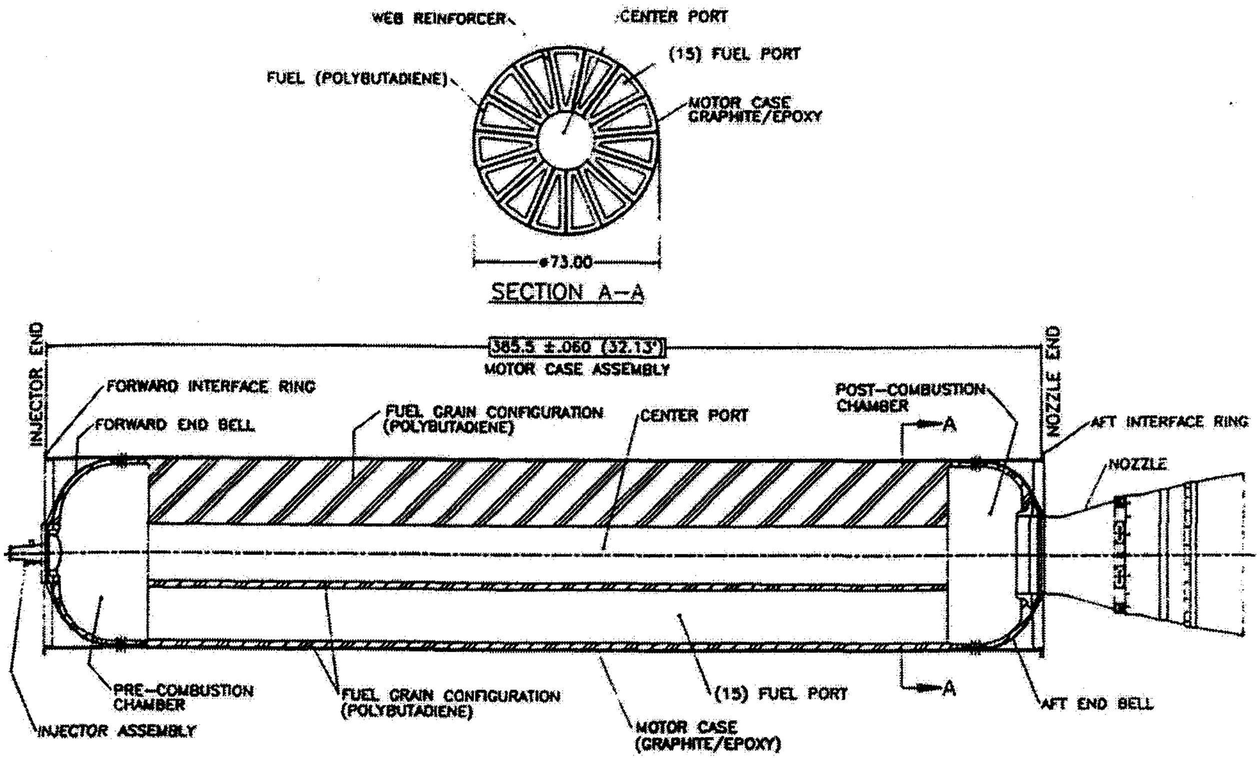


Figure 4 AMROC DM-01 250K-lb$_f$ Hybrid Motor[8]

Table 2 H-250K-lbf Design Parameters[8]

Average Vac Thrust (lb$_f$)	257,000
Avg Vac Specific Impulse (sec)	280
Total Vac Impulse (lb$_f$ sec)	18,500,000
Ave. Chamber Pressure (psia)	400
Burn Time (sec)	72

H-250K-lbf Thrust Motor Development

The first full scale H-250K-lb$_f$ development motor (DM-01) was designed and produced in just 10 months in 1992. This effort required the development of multiple components: a 74-inch diameter, 386-inch long graphite/epoxy motor case; a 9:1 area ratio silica/phenolic nozzle; a 400,000 lb$_f$ thrust horizontal test stand; composite port molds, motor casting fixtures, fuel mixers, and motor manufacturing procedures. The total project duration, from initial design to the completion of testing, was thirteen months.[8]

Test results

The DM-01 motor was tested in a series of four static firings. The first burn was successfully conducted on January 22, 1993. The second burn was successfully completed on February 17, 1993. The third burn was successfully conducted on March 11, 1993. After the third burn, the exit cone had excessive wear, so part of it was cut off before the next test rather than having it fail during test 4. That lowered the nozzle expansion ratio and the thrust. On 24 March 1993, the fourth burn prematurely ended when the case failed.[8] The motors were relatively stable and performance data is shown in Table 3.

Table 3 AMROC 250K-lb$_f$ DM-01 Test Results[8]

Parameter	Burn 1	Burn 2	Burn 3	Burn 4
Thrust (lb$_f$)	216,900	231,900	215,400	214,800
Fuel mdot (lbm/sec)	357	351	339	310
LOX mdot (lbm/sec)	569	600	619	587
ISP (sec)	234	244	225	239
O/F Ratio	1.59	1.71	1.82	1.89
Chamber Prs (psia)	412	419	378	369
Nozzle Area Ratio	8.33	8.00	7.61	3.70
Throat Area (in^2)	364	381	402	418
Vac Thrust (lb$_f$)	257,000	272,300	255,800	235,200
Vac Isp (sec)	278	286	267	262

The second motor (DM-02) was also developed and fired successfully as part of the Hybrid Technology Option Project[9]. However, the motor was only fired once.

After the first firing of DM-02, AMROC discovered that the motor had problems with the manufacturing processes that formed the case insulation and discontinued the testing on that hardware. With investor monies running low, they began to refocus on smaller scale hybrids to investigate fuel and combustion issues. AMROC found it more cost effective to test the motors at Stennis Space Center than at Edwards or their own test facilities[10].

AMROC 10 K-lb$_f$ at Stennis Space Center

AMROC used 10,000 pound thrust liquid oxygen/polybutadiene hybrid rocket motors for research and development work. A number of these motors were tested at NASA's Component Test Facility at Stennis Space Center. These motors have led to advances in combustion stability and material selection for use on AMROC's 250,000 pound thrust hybrid rocket motors. Among the demonstrations conducted there was a pump fed hybrid motor. AlliedSignal Aerospace had been developing high reliability low-cost cryogenic turbopumps based on their foil bearing technology. This was the first application of a turbopump fed hybrid rocket motor. The AlliedSignal foil bearing LOX turbopump and AMROC's hybrid rocket motor were brought together and tested at Stennis Space Center in October 1994 with the first test of the pump fed hybrid rocket motor in November 1994.[10]

Table 4 10K-lbf Turbopump Motor Configuration[10]

Parameter	Value
Fuel Formulation	"DM-02"
Max Chamber Pressure	500 psia
Initial Oxidizer Mass Flux	0.5 lbm/sec/in^2

Injector Pressure Drop	90 psid
Post Combustion chamber L*	250 inch
Pre Combustion chamber length	20 inch

The hybrid motor consisted of a HTPB fuel cast directly into the motor case in a "double-D" two port configuration (Figure 5 and Figure 7). Some of the motor design parameters can be found in Table 4. The fuel was composed of HTPB, an isocyanate curative, and small quantities of additives to achieve the desired mechanical properties. The injector was a showerhead configuration. Some of the stable test data is shown in Figure 8.

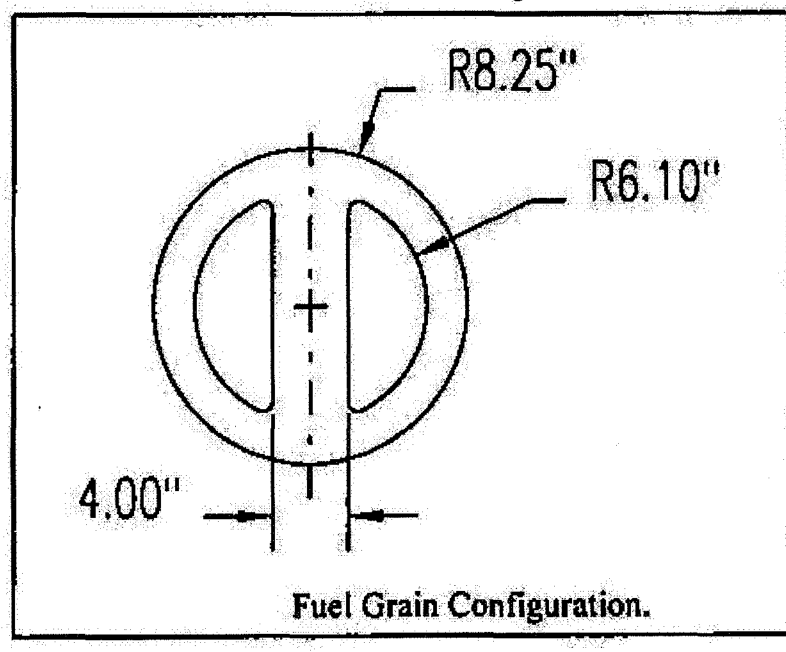
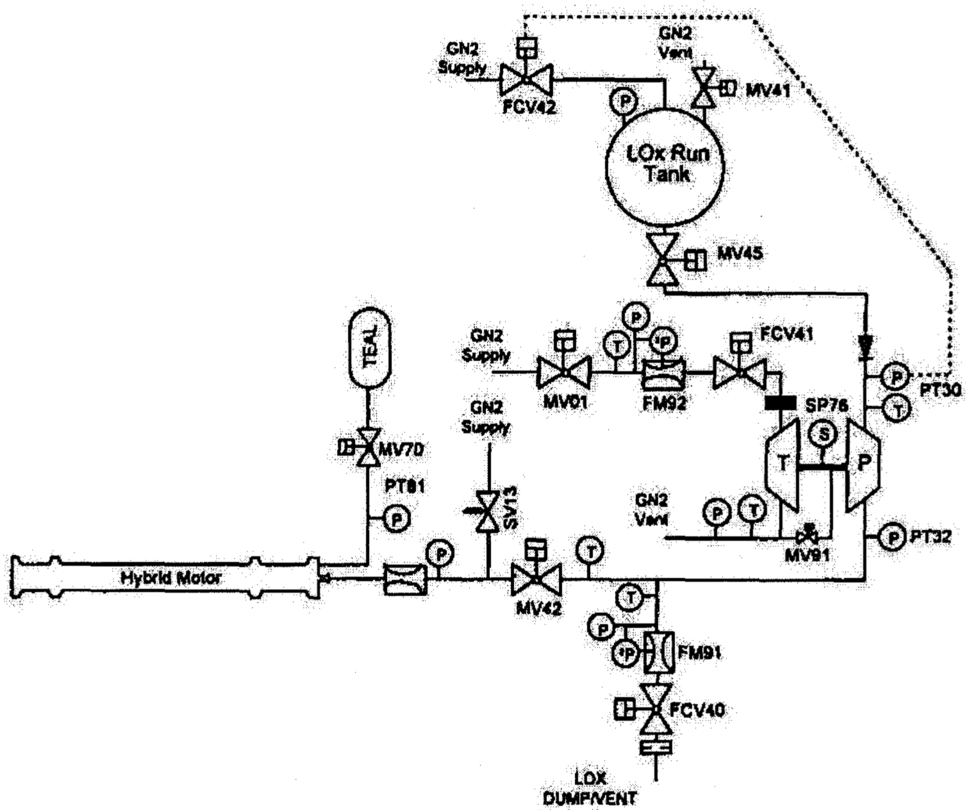


Figure 5 AMROC 10K-lb$_f$ Double D Motor[10]

The test stand configuration for the motor (shown in Figure 6) included a cavitating venturi. The cavitating venturi in Figure 6 is the unlabeled device between the hybrid motor and the valve designated MV42. A cavitating venturi is a proven critical piece of hardware to isolate the hybrid motor oscillations from the liquid oxidizer feed system, ensuring no oscillations on pressure fed systems.[7] These tests, while demonstrating a turbopump and the hybrid motor together for the first time, did not prove that a cavitating venturi was necessary for a turbo pump driven system – it may be desirable from the turbopump operation conditions. A cavitating venturi requires a certain minimum pressure drop across it, and while a turbo pump could probably provide that pressure rise, the pump might be smaller if that extra pressure rise was not required. However, in a turbo pump driven system it may be better to have a steady back pressure behind the oxidizer pump, provided by the cavitating venturi, to keep the changes in motor pressure from affecting the speed of the pump. Without a cavitating venturi between the turbopump and the motor, as the motor pressure dropped over time, the turbopump would tend to spin faster with the reduced back pressure, increasing the oxidizer flow or requiring greater system control to keep it from spinning more.



CTF Pump-Fed Hybrid Fluid Schematic

Figure 6 SSC test arrangement for AMROC turbopump[10]

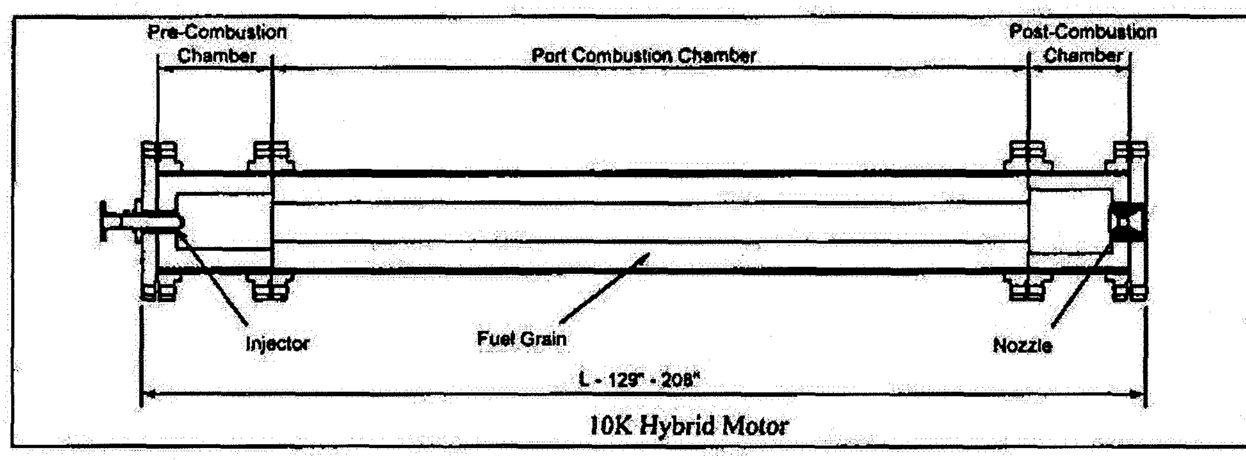


Figure 7 AMROC 10K-lb$_f$ Hybrid Motor[10]

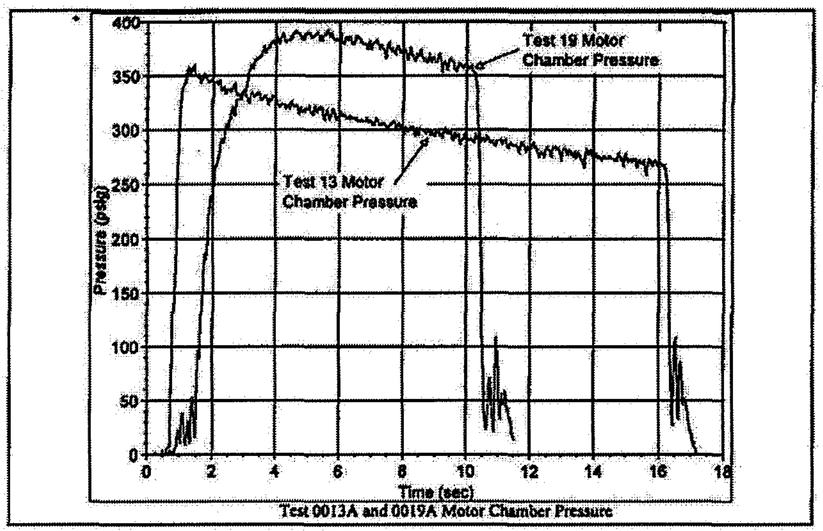


Figure 8 AMROC Turbopump data[10]

Hybrid Propulsion Demonstration Program 250K-lb$_f$ Hybrid Motor

The Hybrid Propulsion Demonstration Program (HPDP) program was formed to mature hybrid propulsion technology to a readiness level sufficient to enable commercialization for various space launch applications.[11,12] Participants in HPDP have included Allied Signal Aerospace, Boeing – Rocketdyne Division, Environmental Aerospace Corporation, Lockheed Martin, Thiokol, United Technologies Corporation – Chemical Systems Division and NASA (MSFC and SSC). The goal of the HPDP was to develop and test a 250,000 pound vacuum thrust hybrid booster in order to demonstrate hybrid propulsion technology and enable manufacturing of large hybrid boosters for current and future space launch vehicles. The HPDP has successfully conducted four tests of the 250,000 pound thrust hybrid rocket motor at NASA's Stennis Space Center.

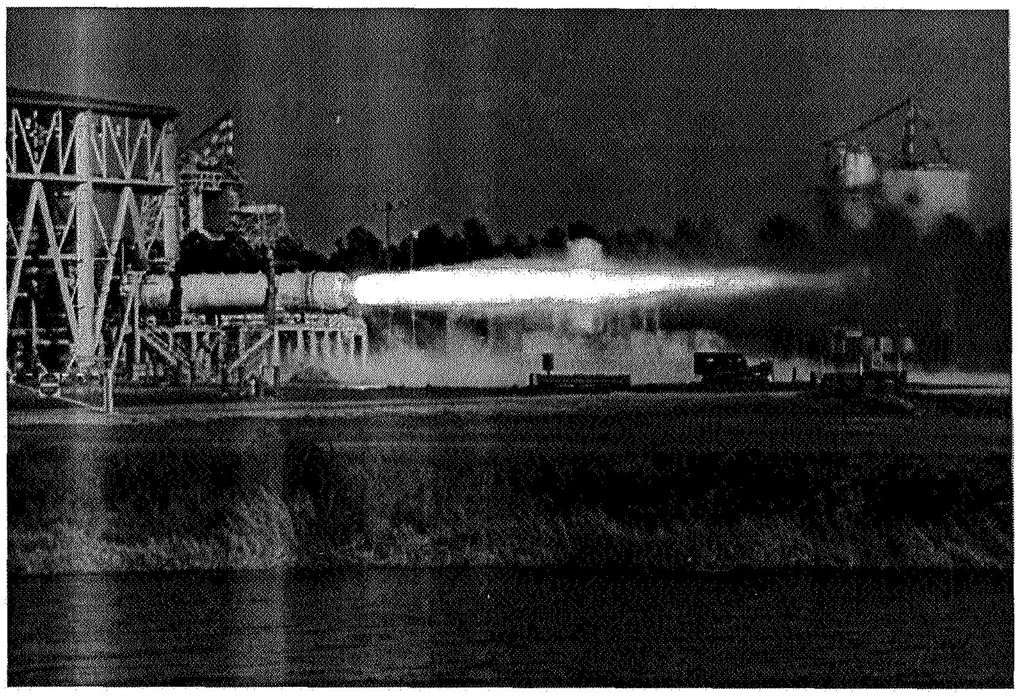


Figure 9 HPDP 250 K-lb$_f$ Thrust Motor 2 Test 3[3]

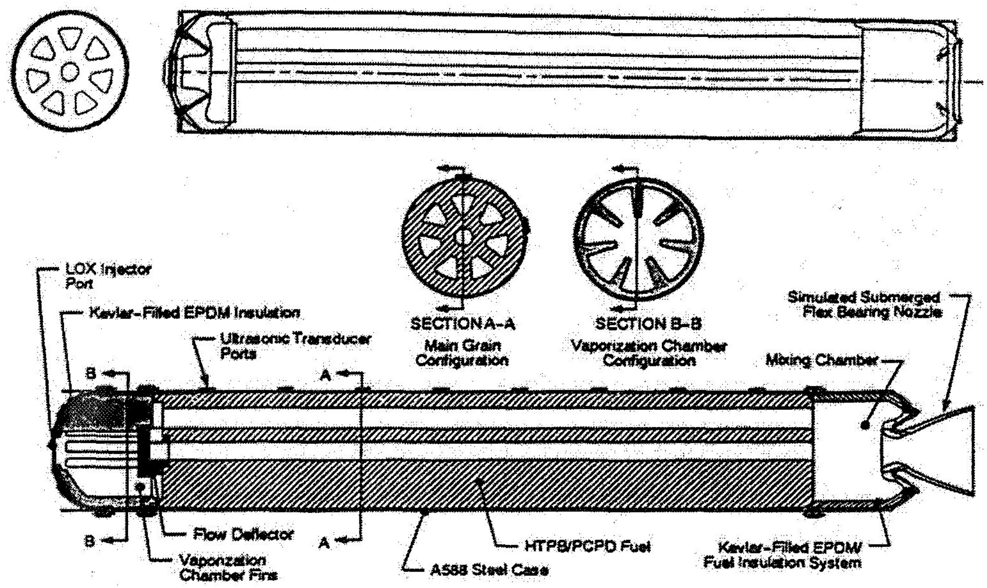


Figure 10 - 250K-lb$_f$ HPDP Hybrid Motor Layout[3]

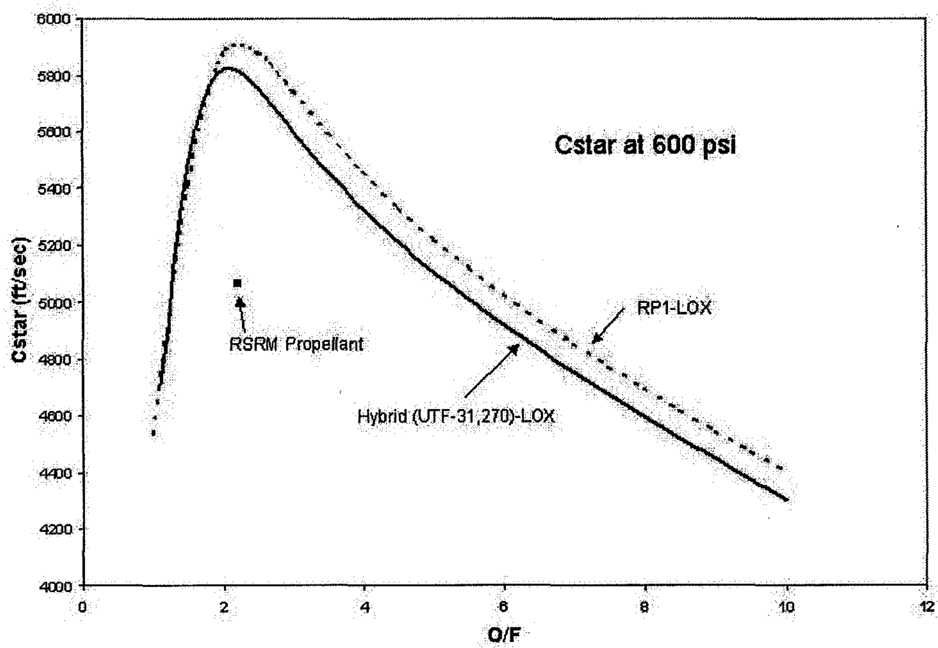


Figure 11 Theoretical Cstar Performance of UTF-31,270 with LOX[3]

11-inch and 24-inch Motor testing at MSFC

There has been significant testing of hybrid motors at MSFC in the 11-inch and 24-inch diameter with GOX and LOX. That testing, from the JIRAD program, the Large Scale Solid Rocket Combustion Simulator program and other programs all fed into the HPDP program.[13,14,15,16] This subscale work continued during the development of the HPDP 250K-lb_f thrust motor and provided the basis for many of the design features of the larger motor.

250K-lb_f LBF Thrust Hybrid Test Motor

250 K-lb_f hybrid motor design requirements are shown in Table 5. Details of the injector, fuel grain, nozzle design are given in references 11 and 12. A photo of a pretest aft end of the grain is shown in Figure 12.

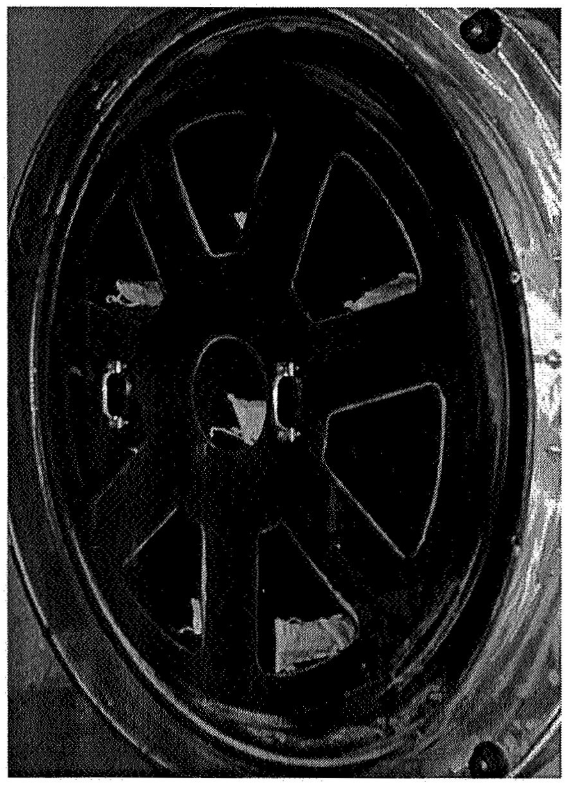


Figure 12 250K-lb$_f$ HPDP Motor Ports[3]

Table 5 250K-lb$_f$ Design Parameters[3]

Parameter	Value
Max. Vacuum Thrust	250,000 lb$_f$ (1 112 055. newton)
Ave. Vacuum Specific Impulse	280 sec
C* Efficiency	98%
Max. Operating Pressure	900 psia (61.2 atmosphere)
Ave. Chamber pressure	750 psia (51.0 atmosphere)
Burn Time	80 sec
LOX Flow Rate	600 lbm/sec(272. 2 kilogram/sec)
Oxidizer Flux Level	0.64 lbm/sec/in^2(0.045kg/sec/cm^2)
Port Length	380 inch (9.65 meter)
Length to Diameter Ratio	35.3
Fuel/Oxidizer	HTPB*/LOX

* HTPB and polycyclopentadiene (PCPD) with no metal additives

Head-end designs

In order to address the combustion stability concerns that had been found in the development of large scale hybrid rockets,[7,17,18] the HPDP consortium came up with two ways to try to control the combustion instability: a passive technique, with no moving parts (employed on Motor 1) and an active approach, utilizing heat addition from the forward end (employed on Motor 2).

Motor 1 Design Basis and History

The Motor 1 head design was based upon previous solid fuel ramjet stability historical data, with the creation of 'a stable zone of hot, recirculating, combustion gases ahead of the establishment of the primary combustion zone.'[19] Several of these concepts were designed and subscale tested with gaseous oxygen (GOX) in the 11-inch diameter motor, which was ignited with an oxygen hydrogen torch. 'All oxidizer dump plenum configurations that produced flow recirculation of combustion gas at the leading edge of the diffusion flame sheet resulted in stable operation. Configurations that did not produce such flow structures exhibited unstable combustion.[19]' Testing with LOX in the 11-inch diameter hybrid motor produced similar stability results. 'The comparison showed that flow field features which reduced or eliminated acoustic oscillation in motors using gaseous-oxygen injection were also required to stabilize combustion in liquid-oxygen-injected motors.' This testing evaluated the effects of short and long fuel lined vaporization chambers with different flame holding concepts – fuel fin, flame holder and fuel inhibitor. The LOX was injected with either a solid cone or axial injector. The combination of the long fuel lined vaporization chamber with the fuel fin and the solid cone injector had the lowest average non-acoustical oscillation amplitude percentage and also had the highest vaporization chamber heat output. It was determined by the authors that 'Fuel fins are effective in both short and long vaporization chambers in reducing the average instability level associated with liquid-oxygen injection. This is most likely because of the combustion port, spanwise, hot-gas recirculation zone behind the fuel fin and flame from combustion on the fin surface entering the combustion port. Incorporation of fuel fins appears to offer a viable means of scaling a combustion oscillation suppression method to larger liquid oxygen based motors.[20]' This work supported the concept that became the bases for the design of motor 1's head-end.

Testing with the 24-inch diameter LOX motor was started in parallel with the 11-inch diameter GOX motor testing to support the ramjet combustion stability concepts.[21] Testing was conducted with domed shaped vaporization chambers with varying length fins and no fins. This published reference[22] had no conclusions listed on the effect of fins; however it did support previous American Rocket Company (AMROC) conclusions stating that the oxidizer feed system must be decoupled from the motor oscillations. This decoupling was implemented by moving the cavitating venturi, which regulates the liquid oxygen flow, from well upstream of the injector to right before the injector. A well designed cavitating venturi speeds the fluid to the point where the local fluid pressure is less than the vapor pressure of the fluid and the fluid flashes, and then the flow rate is controlled by the vapor pressure, not the downstream pressure. This effectively eliminated the feed system coupling of the oscillations with the motor oscillations[7]. Subsequent HPDP testing of the 24-inch hybrid motor evaluated the effect of the center port on combustion efficiency.[16,23] These tests showed the effect of the center port on the motor combustion, by blocking the center port with a fuel plug or making a tortuous path to the center port by use of a fuel flow port deflector. Blocking the center port lead to more stable motors compared to unblocked center port motors, but the center port open motors were within the +/-2.5 % stability band HPDP requirements. The +/-2.5 % stability band was an indicator of the stability, based on the pressure variations verses a 1 second moving average of the low speed chamber pressure. The final motor of that 24-inch diameter series incorporated a flat-topped fuel flow deflector, a fuel lined vaporization chamber with fins and a nozzle throat designed to provide a chamber pressure of 900 psi. This configuration of motor 'showed that altering conditions in the center port provided a more stable motor with high combustion efficiencies. Results from the incorporation of the fuel flow deflector also indicate that a more uniform regression along the length of the grain was obtained. These data resulted in the incorporation of the fuel deflector into the first 250 K-lb_f motor.'[16]

Motor 2 Design Basis and History

Motor 2's head-end design was also influenced greatly by historical data, initially being based upon data from the American Rocket Company (AMROC). During the late 80's and early 90's, AMROC was the leader in hybrid technology. Some of their combustion stability experience is listed in a patent[24] and a paper on combustion stability[7]. Based on AMROC's published documentation, hybrid combustion instability was thought to be caused by several reasons, 'One of the causes of erratic performance is the flow of unvaporized liquid oxidizer, which disrupts the normally stable boundary layer combustion process. Ideally, during combustion a combustion zone is formed in the boundary layer at the interface of the vaporizing fuel flow and the vaporized oxidizer, within the momentum boundary layer and is the source of the heat flow to the surface of the solid fuel to maintain fuel vaporization. As unvaporized liquid oxidizer is distributed along the surface of the solid propellant (grain), the temperature of the forward reaction mixture is reduced, thus the efficient combustion area is developed toward the aft end of the rocket. As the pressure differences within the combustion area increase, the hot reaction products move forward into the area of low pressure and temperature, then aft again, producing a series of low frequency oscillations along the length of the grain. This results in erratic combustion and unstable thrust. Thus, it is essential for stable hybrid rocket engine performance that there is a consistent boundary layer over the entire solid propellant.[24] Another large cause of combustion stability AMROC documented included feed system coupling with the hybrid combustion – this they addressed by a cavitating venturi just upstream of the LOX injector[7]. AMROC's suggested correction of the boundary layer problem is to inject a phyrophoric liquid into the oxidizer stream to vaporize the oxidizer before entry into the combustion zone. 'The hypergolic fluid is injected in an amount sufficient to vaporize all of the liquid oxygen. The flow rate can be readily calculated from the temperature of the liquid oxidizer and the flow rate of the oxidizer. For example, a hybrid engine using liquid oxygen and a trialkyl aluminum pryogolic fluid, a flow rate of about 0.1% by weight of the liquid oxidizer is sufficient to vaporize all the oxidizer. Flow rates higher than 5% by weight of the oxidizer are unnecessary and can lead to unstable burning. Usually the flow rate is from about 0.5 to 3.0% by weight of the oxidizer.'[24]

To support the claims in AMROC's patent, they included test data from a series of tests, using the configuration shown in Figure 13. 'Hybrid engines were constructed incorporating a polybutadiene solid grain and utilizing a casing containing a precombustion zone as shown in [Figure 13]. Liquid oxygen was utilized as the liquid oxidizer and triethyl aluminum (TEAL) as the hypergolic fluid. One engine (Example 1)[Figure 14 H8#1] was operated with TEAL only injected during initial start ups. Two other engines (Example 2 and 3) [Figure 14 H8#2 and

H8#3] were operated with the TEAL injected continuously. Example 4[Figure 15] was a test burn lasting 70 seconds with TEAL continuously injected. *Figure 14* shows three short test firings; Example 1[Figure 14 H8#1] shows the aft port pressure during a time when TEAL was not injected. Both Example 2 and Example 3 [Figure 14 H8#2 and H8#3] show the aft port pressure, under identical conditions, while TEAL was being injected. Example 1 shows the low frequency harmonics (oscillations) of hybrid rocket engines that have been reported in the literature while Example 2 and 3 show that said low frequency harmonics have been eliminated.'[24]

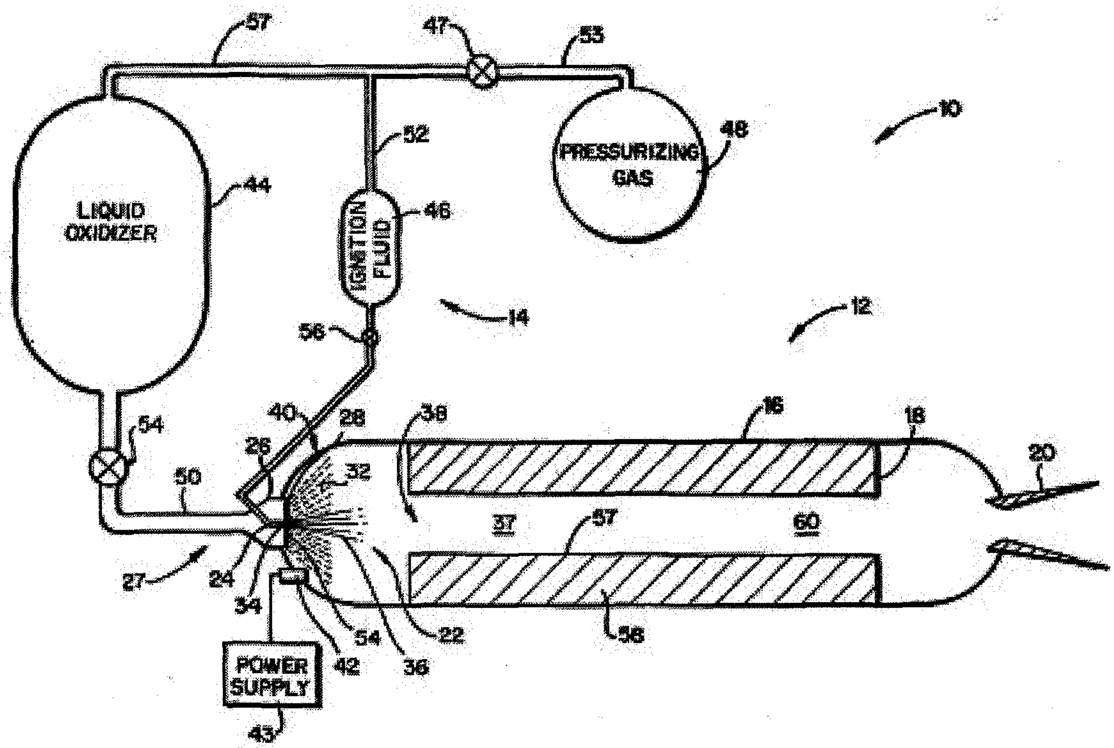


Figure 13 US Patent 5582001 Motor Layout (32-LOX spray, 34-ignition fluid injector, 36-ignition fluid spray, 42 LOX injector)

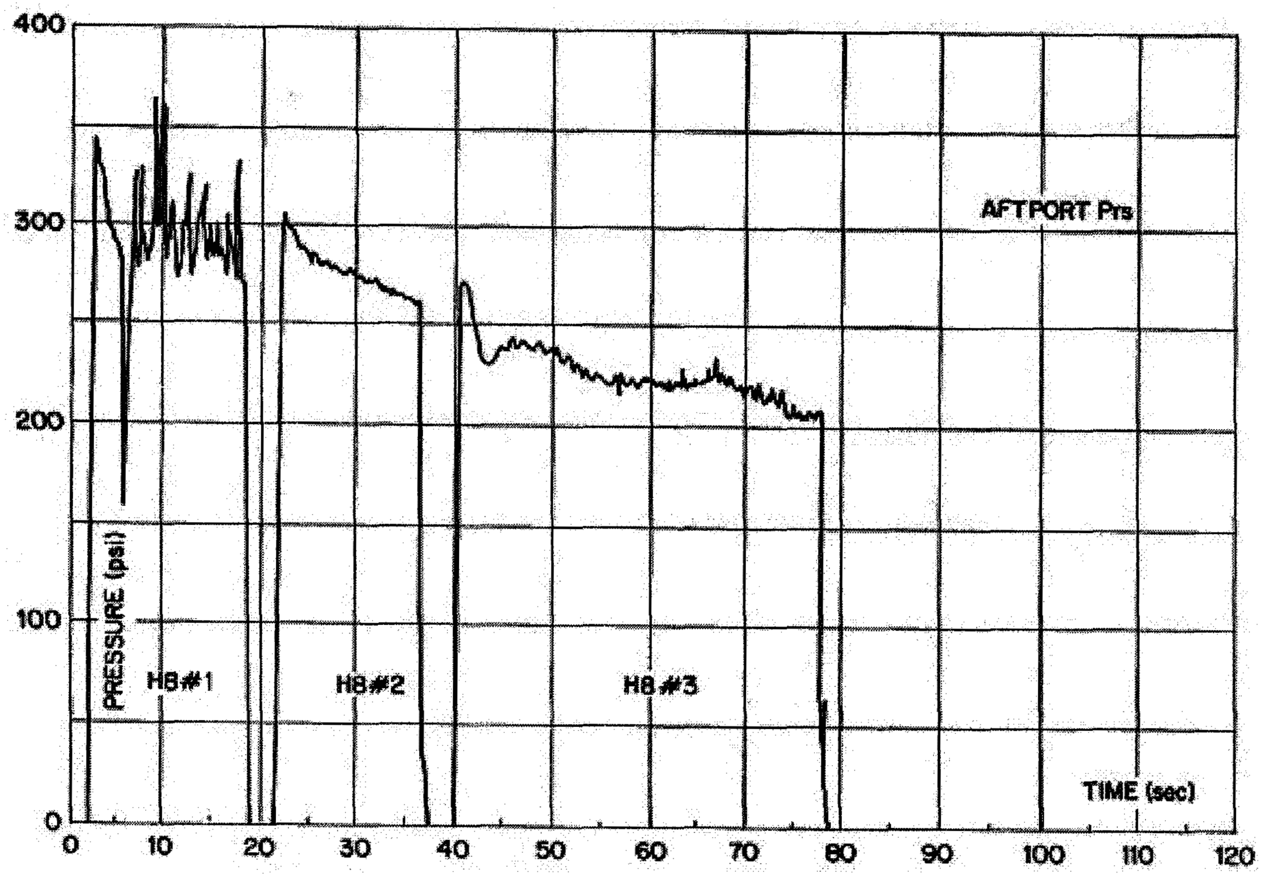


Figure 14 US Patent 5582001 Motor Plot 1-Effect of TEAL Addition on Chamber pressure(H8#1 – no Teal, H8#2 and H8#3 TEAL on)

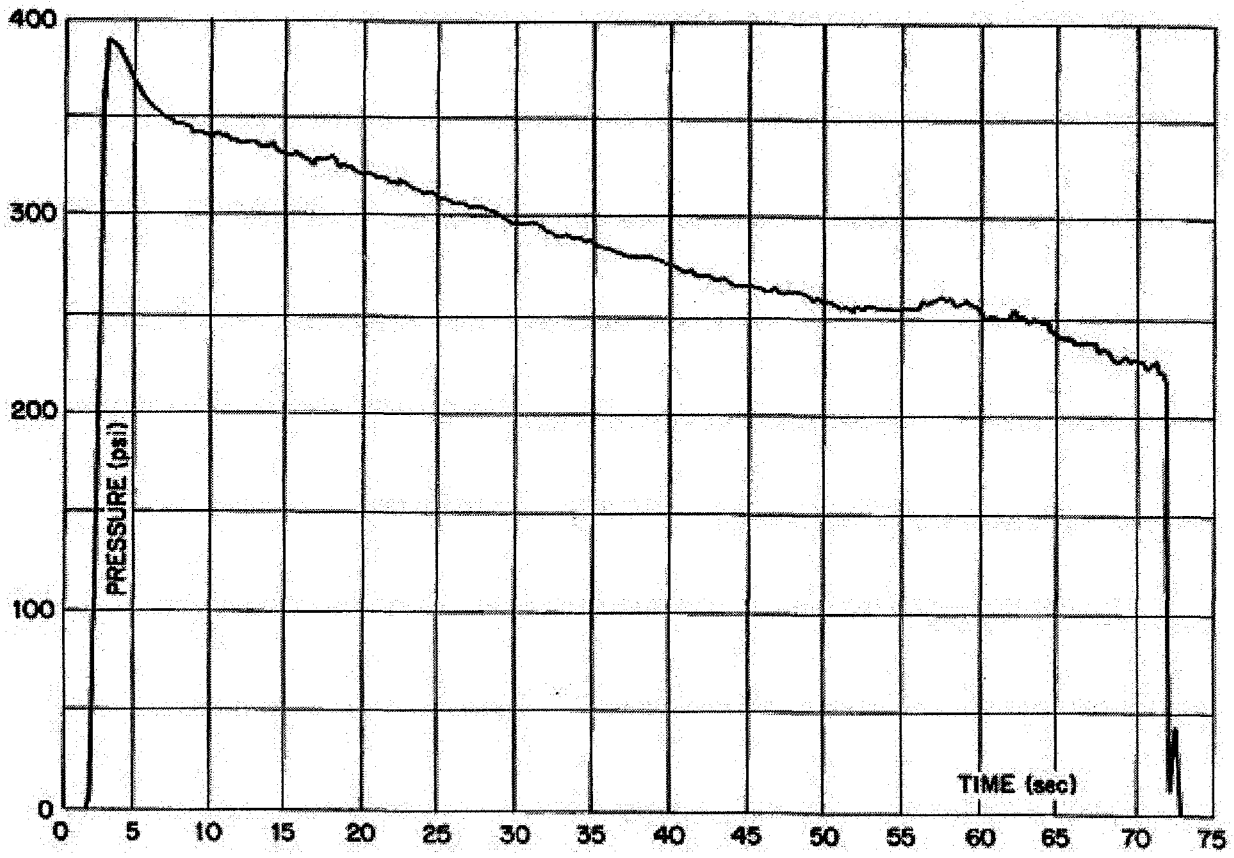


Figure 15 US Patent 5582001 Motor Plot 2-TEAL flowing the entire burn

These AMROC conclusions were also somewhat supported by HPDP testing with LOX on the 11-inch motor. This series of motors were ignited by TEA/TEB, which is a mixture of pyrophoric liquids. 'The heat input from the TEA/TEB combustion is approximately 3,600 Btu/sec, substantially exceeding that available from steady-state combustion of fuel in the vaporization chamber. Heat required to vaporize liquid oxygen is approximately 90 Btu/lbm, or approximately 400 Btu/sec at average motor liquid-oxygen flow rates. Thus the heat available from TEA/TEB combustion is well in excess of that necessary to vaporize the liquid oxygen. Examination of motor pressure data indicates that the effect of TEA/TEB combustion on stability appears to be significant in some cases and less so in others.'[20]

Motor 2's design (and Motor 1's also) came from the inherent safety consideration and argument between a safe, solid inert hybrid fuel with a zero TNT equivalency requiring a pyrophoric liquid to start it and make it operate in a stable condition. The special care needed to handle the pyrophoric liquid had raised a question of the handling safety of the whole system. Also, members of the consortium had come to doubt if Motor 1 would be stable, so it was decided to test a concept where the ignition and the stabilizing heat would come from a hybrid motor itself instead of a pyrophoric liquid. Motor 1 retained the pyrophoric liquid for ignition purposes only.

Additional testing was conducted on the 11-inch and 24-inch diameter LOX motors to evaluate this concept. Reference 16 discusses the results of the 11-inch motor ignition system and the 24-inch motor vaporization system testing. Testing of both sized motors was performed to see if small hybrids could start large hybrids and if the heat could keep it stable. Startup was smooth and combustion stability was increased compared to motors without this active heat source. The success of this testing led to the incorporation of the hybrid "heater motors" into the 24-inch Large Subscale Quad Port Test Series. "On test HP24-8020, the (GOX feed to the heater motor) system was terminated at T+11 seconds which caused the motor to go unstable.... The test confirmed the hypothesis, as shown in multiple 11- and 24-inch tests series, that the flame anchoring in the head-end of a hybrid motor is essential for motor stability."[16]

The conclusion that heat addition was necessary was also supported by the 11-inch diameter GOX testing which was looking for passive techniques for combustion stability. An interesting footnote to that work was the conclusion that 'heat released from combustion of hydrogen gas in the dump plenum at an estimated mixture ratio of 120 also stabilized combustion in configurations that were otherwise clearly unstable.' This conclusion was also used in the design and development of Motor 2.[4]

Two HPDP tests published in reference 23 show the effect of fins and no fins on multiport hybrid motors. These motors were tested with the same conical injector. The motor having fins in the forward dome had more fuel regress (19.64 lbm) than that of the motor dome without the fins (4.24 lbm), even if corrected for the burn time differences (~18 vs ~8 seconds). However, that additional head-end fuel regression did not result in an increased motor C* efficiency (both yielded 98%) or combustion stability, as judged by the chamber pressure average oscillation divided by average pressure (1.60% vs 1.60%). The conclusion that can be drawn from tables 4, 6 and 12 of reference 23 is that the impingement of LOX on the head-end fuel fins can cause it to erode, but that additional fuel flow may not contribute to combustion stability or an increased C* efficiency.

The Motor 2 head-end design that was eventually built and tested was similar to the patented design[25] (Figure 10 and Figure 16). It incorporated heater motors to start the main motor and provide heat to vaporize the LOX for combustion stability. An axial injector was designed for Motor 2 since Motor 1 used a conical spray pattern injector, and AMROC's patent[24] indicated that LOX impingement on the burning fuel surface could be a cause of the instability. The head-end also incorporated a recirculation area in the front end, where gaseous oxidizer would theoretically recirculate and burn the head-end fuel, generating even more heat.

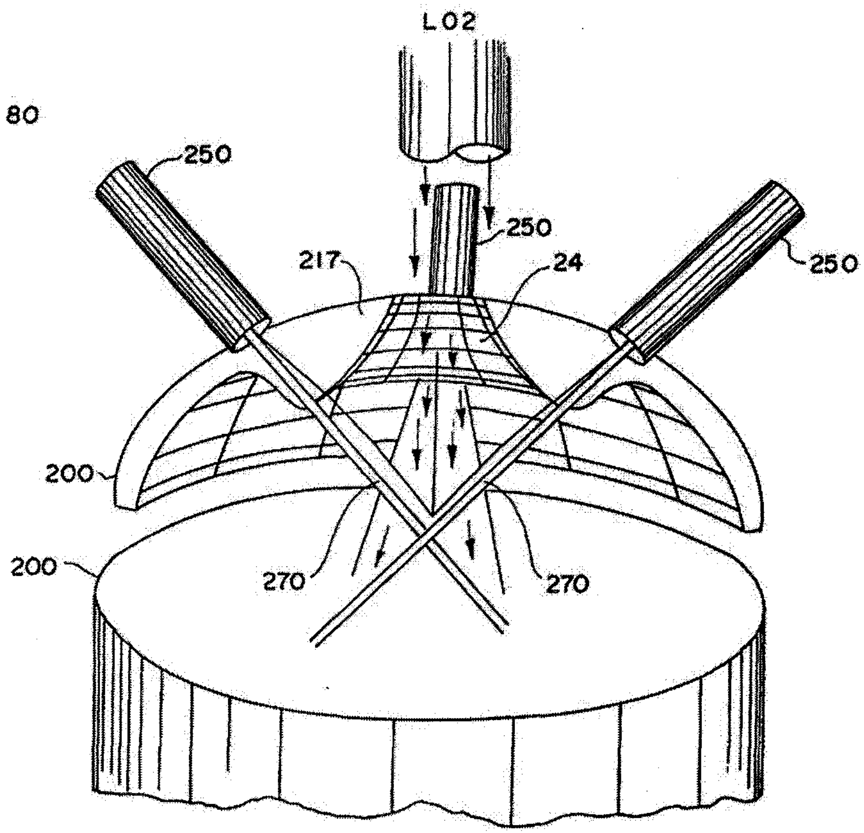


Figure 16 US patent 5794435 - Stable-combustion oxidizer vaporizer for hybrid rockets (200-hybrid fuel, 250-small hybrid motors, 270-small hybrid motor exhaust gas flow, the arrows are LOX flow)

Ballistic Tests

Motor 1 Test 1

Motor 1 test 1 was the first of the 250K-lb_f hybrids tested at Stennis Space Center and was conducted on July 9, 1999. This was the passive combustion stability design employing fins in the headend and a flow deflector over the center port (Figure 10). It was lit by TEA/TEB and exhibited unstable behavior (Figure 17). Due to an external TEA/TEB system fire, the test conductors terminated the test prematurely. There was minor scorching of some of the TEA/TEB ignition system components, however no damage to the test stand. Calculations have shown that the requested TEA/TEB flow rate to motor was supplied even though some TEA/TEB escaped to the atmosphere. Subsequent testing of the TEA/TEB ignition system indicated failed pressure transducer diaphragms, which were over pressured due to water hammer effect causing the TEA/TEB to leak. Once the TEA/TEB, a phyrophoric liquid, came in contact with air, it burned.

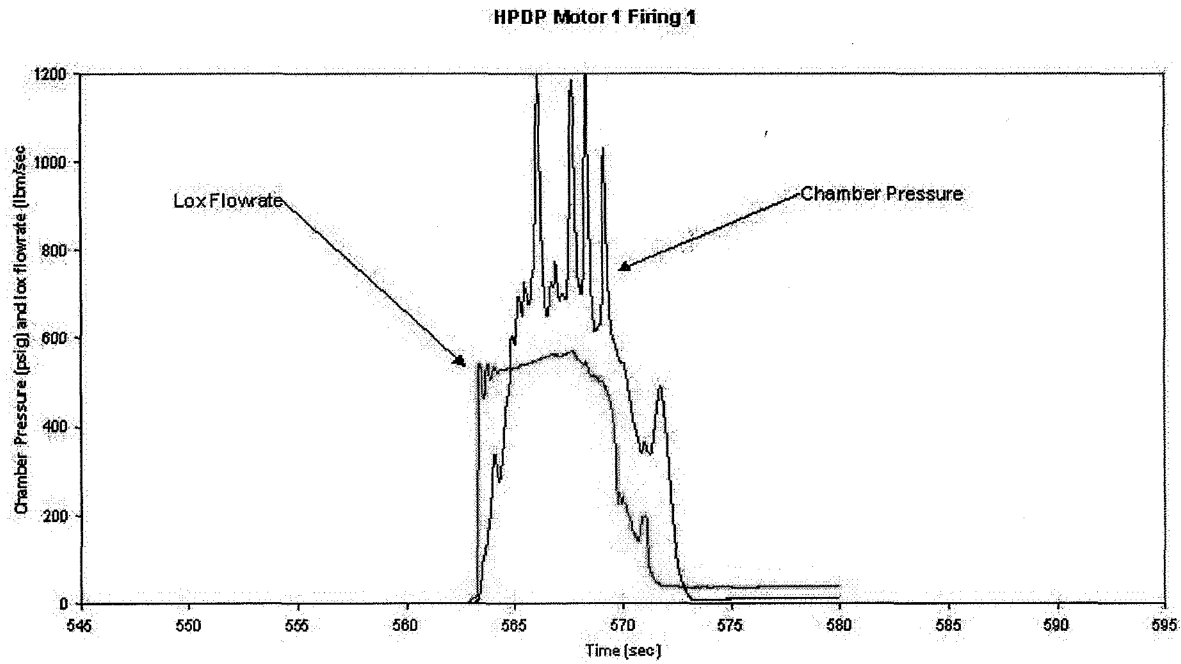


Figure 17 HPDP 250K-lb$_f$ Motor 1 Firing 1[3]

Motor 2 Test 1

Motor 2 Test 1 was the first test of the active combustion stability system, with embedded heater motors in the head-end. The ignition system consisted of two banks of small gaseous hybrid motors embedded in the forward dome of the motor. The test was conducted on August 13, 1999. Ignition was smooth and combustion was stable (Figure 18). A small pressure blip that occurred during the first few seconds of the test was believed to be from the backlighting of one bank of the gaseous hybrid motors in the head-end. Pretest checks indicated that the ignition system of one of the banks of gaseous hybrid motor was shorted out.

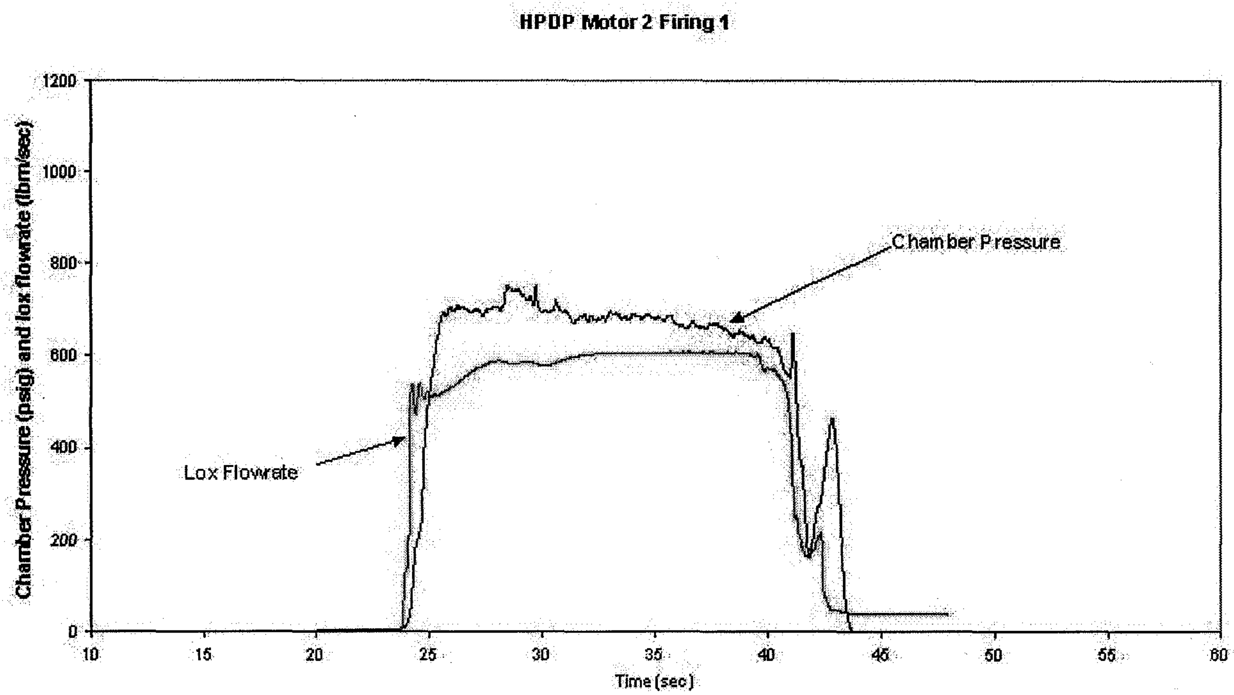


Figure 18 HPDP 250K-lb$_f$ Motor 2 Firing 1[3]

Motor 2 Test 2

Motor 2 Test 2 was a refiring of the Motor 2 Test 1 hardware, except the nozzle from Motor 1 Test 1 was used. The test was conducted on September 9, 1999. The nozzles, by design, were refurbished between each test and the nozzle from Motor 1 test 1 was available and had eroded less than the nozzle from Motor 2 test 1.

Motor 2 Test 2 ignited smoothly, however large pressure oscillations were encountered during the burn (see Figure 19). It is believed the small gaseous hybrid heater motors, as they burned (the ports got bigger and the flux dropped which shifted the O/F), produced less heat to provide the amount necessary for LOX vaporization and for holding the flame at a fixed location for establishing for combustion stability.

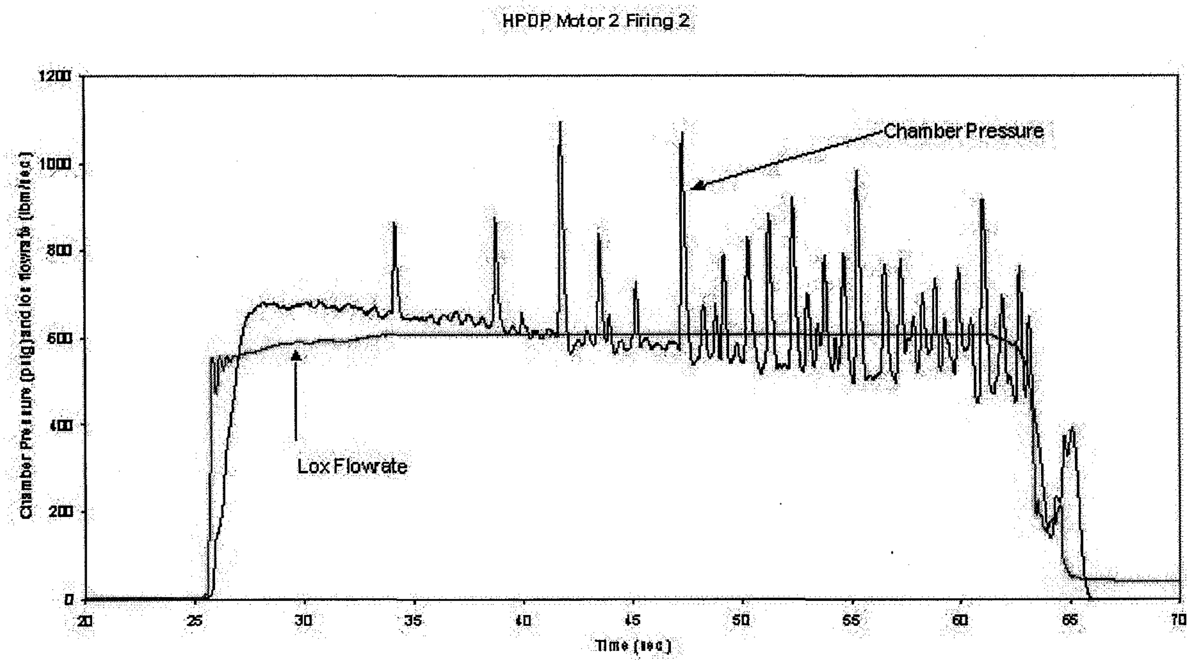


Figure 19 HPDP 250K-lb$_f$ Motor 2 Firing 2[3]

Motor 2 Rework

Since the small gaseous hybrids for heater motors had burned till they were no longer able to provide a sufficient heat source and/or flame holding device, they were drilled out and recast in a slightly different configuration.

Motor 2 Test 3

Motor 2 Test 3 was reassembled using the refurbished nozzle from Motor 2 Test 1. The test was conducted on January 17, 2002 and exhibited a smooth ignition and steady pressure trace (see Figure 20). The small pressure disturbances/blips are believed to be from ejecta. Part of the recasting of the head-end were found post test outside the motor.

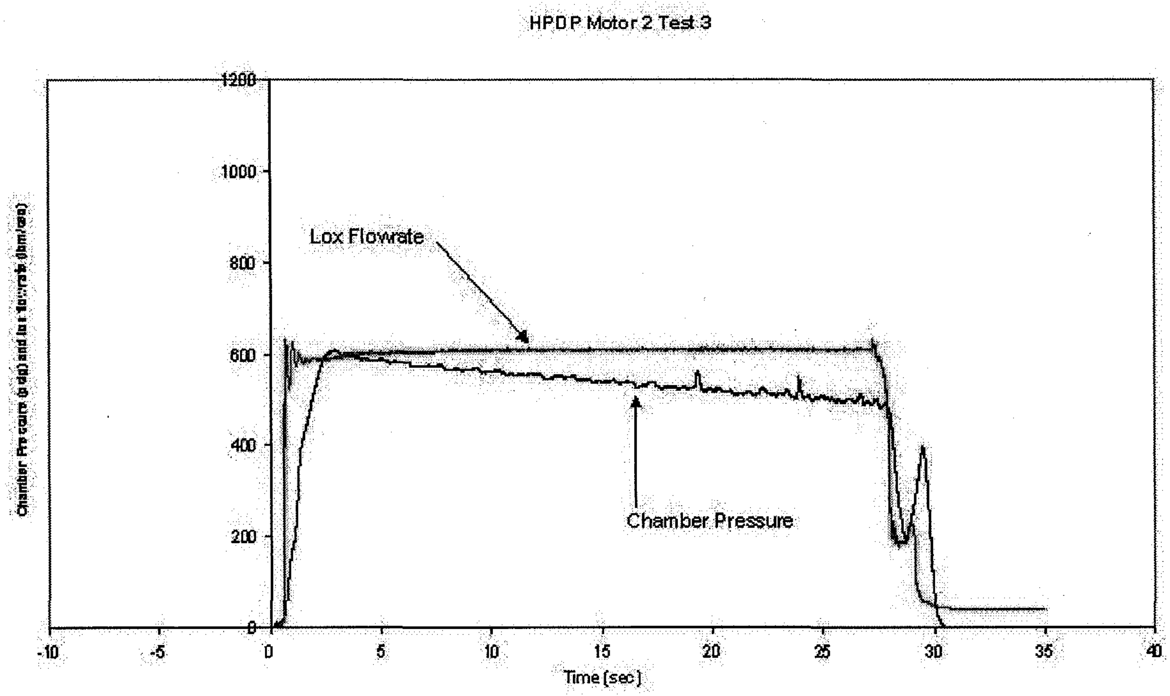


Figure 20 HPDP 250K-lb$_f$ Motor 2 Firing 3[3]

Motor weights were calculated by three techniques during the 250K-lb$_f$ program. The first technique was to weigh the components or sometimes the assembled motor on truck scale (at MSFC and/or SSC). The second technique was system called the bore crawler. It used mechanical arms and fingers to measure the port geometry pre and post test. Data from that technique was published in a paper on the 250K-lb$_f$ hybrid[26]. A third technique was developed that used a laser to map the port area. The laser was pulled thru the individual ports pre and post test and area of the ports at those locations were calculated. From that the motor weights were calculated. The data from the laser technique, indicating the port shape, can be seen in Figure 21.

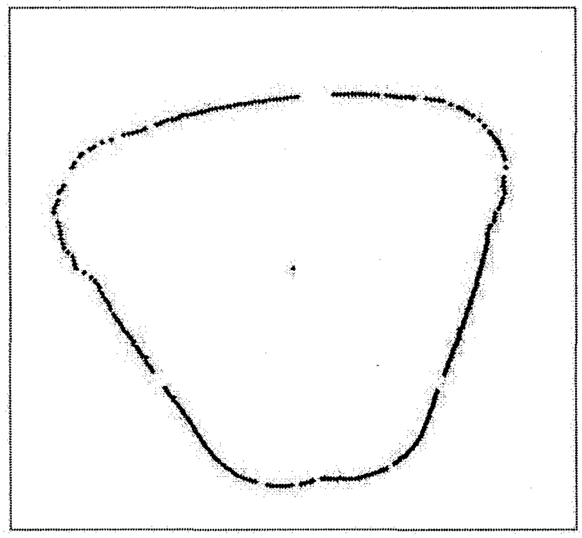


Figure 21 Laser Port Mapping Sample – Pretest Port[3]

Average regression rate data the ports per test can be shown in Figure 22, Figure 23 and Figure 24. There was a significant difference between the three weighing techniques, with the maximum percentage differences of techniques near 10%. This has lead to some uncertainty in the performance calculations. Another possible contributor to the uncertainty in the performance calculations is that the cavitating venturi was never calibrated.

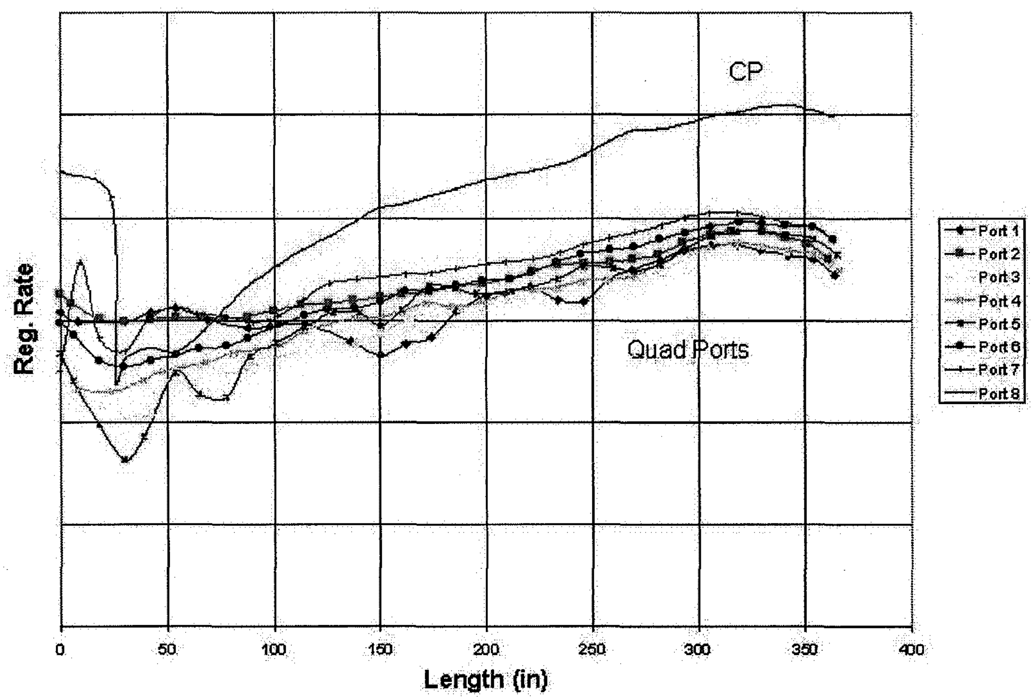


Figure 22 HPDP 250K-lb$_f$ Motor 2 Test 1 Regression Rates(Center Port and Quad Ports)[3]

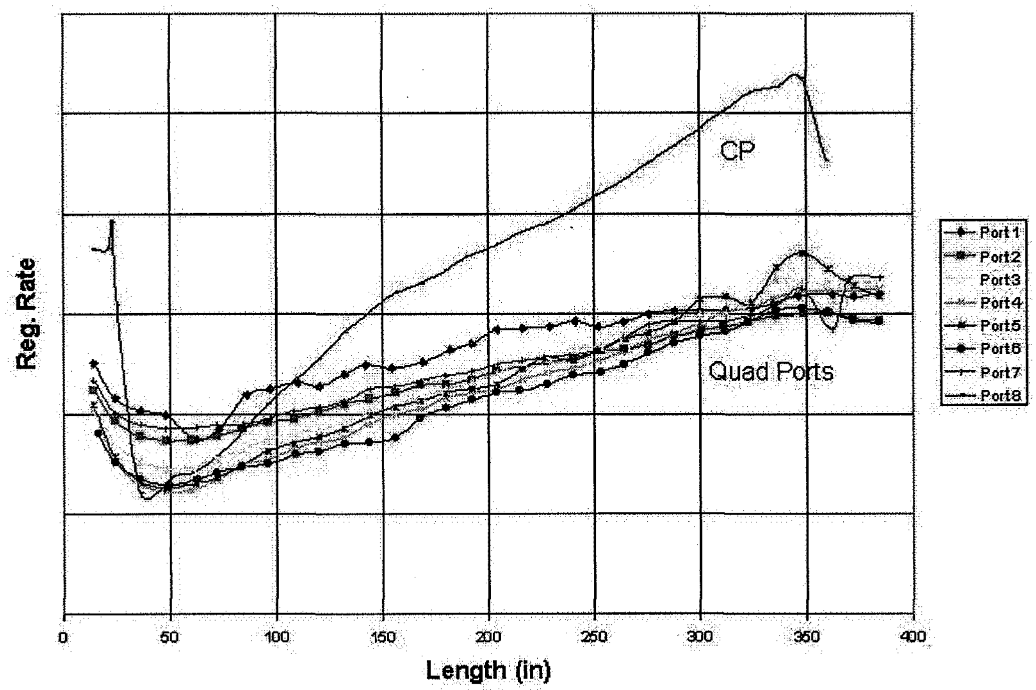


Figure 23 HPDP 250K-lbf Motor 2 Test 2 Regression Rates(Center Port and Quad Ports)[3]

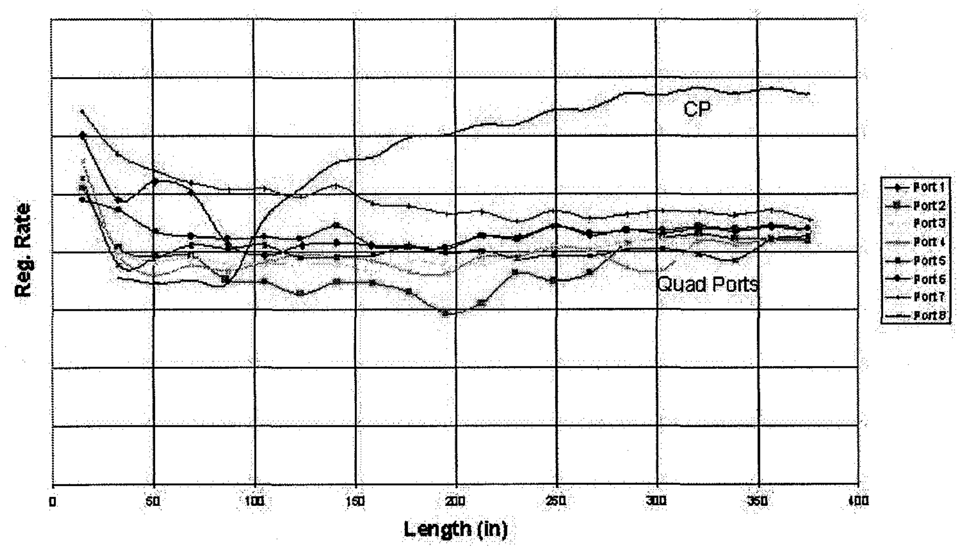


Figure 24 HPDP 250K-lbf Motor 2 Test 3 Regression Rates(Center Port and Quad Ports)[3]

Table 2. Average Motor Performance Parameters[3]

Parameter	Motor 1	Motor 2		
	Test 1	Test 1	Test 2	Test 3
Thrust (lb_f) - Vac	177136.9	186336.9	210065.5	195989.4
ISP VAC	250.0	248.6	276.9	263.9
ISP VAC EFF	0.77	0.78	0.90	0.92

Cstar	4,576.3	4,855.9	5,092.9	5,044.2
Cstar %	78.7	84.9	93.5	97.6
Global O/F	2.3	2.8	3.5	4.5
Duration (sec)	7.9	18.6	38.9	28.0
Chamber Pressure (psia)	594	625	600	542

CSTAR chart from Theoretical calculations with PC=600 psia is shown in Figure 25. The test O/F and ISP/CSTAR calculations are from HPDP final report[27] with Laser mapping of center port weights.

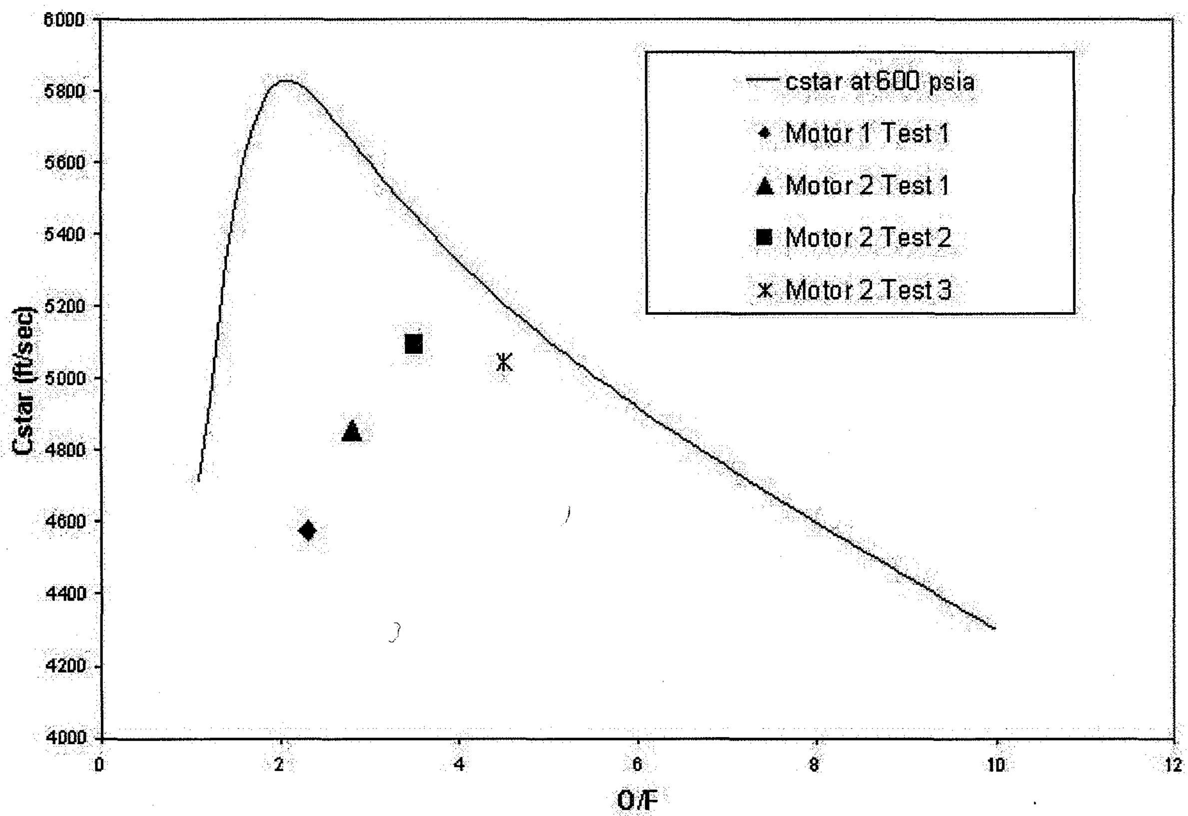


Figure 25 Theoretical Cstar vs Data[3]

Performance Analyses

The global performance calculations for the motors are shown in table 2.

The high O/F ratios for the motor 2 tests can be attributed to two things – scale up from small hybrid rocket motor burnrates and the typical shift in O/F seen in hybrid motors. The 250K-lb$_f$ hybrid was designed based on a motor with a hydraulic port diameter of 2. The hydraulic port diameter of the 250K-lb$_f$ motor ports was on the order of 4 times as large. Subsequent testing of the ¾ scale motor, a large single port quad motor, in the HPDP program provided a clue as to what would happen, an expected 30% reduction in fuel regression rate[2]. Other work comparing small ports regression rates extrapolated to larger ports showed an error in the regression rates greater than 10%[1]. Reduction in the fuel flow rate affects the O/F, chamber pressure and thrust.

Stability of the tests can clearly be seen in the spectrograms of the test data Figure 26, Figure 27, Figure 28, and Figure 29. The spectrograms show the Power Spectral Densities over time, with the amplitude at the right representing the logarithmic magnitude of the oscillations. The tests where the heater motors provided stability are easily recognizable. Another way to look at the stability is shown in Figure 30, Figure 31 and Figure 32, which show the filtered composite normalized by average pressure. The bandpass filter is between 5 and 500 Hz to remove the non-acoustic response. The oscillations upstream of the injector show the noise from the cavitating venturi. The two pressure blips in Motor 2 test 3, which are believed to be ejecta show up quite well. The unstable nature of motor 2 test 2 is also quite evident in Figure 31, with the RMS excursions denoting an excitation in acoustic activity concurrent with the low frequency events.

General

A multiple port grain configuration was used in 250K-lb_f hybrid motors due to the low fuel regression rate requiring a lot of surface area to generate the fuel flow necessary for desired thrust level. The head-end and the aft end attached to the each side of the main fuel grain represent a pre-combustion chamber for heating and vaporizing LOX and a mixing chamber for completing reaction of unburned fuel with oxidizer, respectively. One explanation for the chamber pressure oscillations that occurred in Motor 1 Test 1 and Motor 2 Test 2 may be because of different fuel regression rates in the multiple chambers (quad ports and center port) resulted from uneven LOX distribution, incomplete vaporization of LOX at lower temperature, and not thorough combustion in the mixing chamber. The operation of the heater motors in Motor 2 tests 1 and 3 seems to have corrected for this phenomena. However, incomplete reaction of fuel with oxidizer in the ports and in the aft mixing chamber may have lowered the motor combustion efficiency in all of the motors.

In order to prevent unstable combustion in hybrid motors, flow and combustion conditions under the lower temperature of LOX and very oxidizer-rich environment at the forward end of the fuel grain need to be precisely determined to establish a proper flame front, which keeps the motor stable. A proper flame front was demonstrated using the hybrid heater motors on Motor 2 tests 1 and 3.

Performance

Motor performance in terms of the C* efficiency yields 78 to 97% while in terms of Vacuum ISP yielded reveals 77-92%. The low C* efficiency implies that the fuel that was released from the grain did not burn completely, which may have been due to poor mixing of the oxidizer-rich and fuel-rich areas of the gasses in the motor. The ISP efficiency was lowered by the C* efficiency issues, as well as the low pressure due to the lower than expected fuel regression rate. Based on the bore crawler data, the amount of fuel regression in Motor 1 indicates severe difference from each port[26]. The amount of regressed fuel of the quad ports vary from 155 lbm to 220 lbm with the center port of 112 lbm, which is equivalent to the minimum regression of the quad ports after compensation of the cross sectional area ratio. The low regression of the center port in Motor 1 is believed to be because of the existence of the flow deflector, causing a tortuous path for the LOX to take. In contrast in Motor 2, fuel regression in the center port exceeds the maximum regression in the quad ports, implying a larger amount of oxidizer flowing through the center port than the quad ports. Motor 2's axial injector directs the LOX directly toward the center port.

Pressure

Motor pressure-time characteristics in Figure 17, Figure 18, Figure 19, and Figure 20 exhibited both stable and unstable combustion, especially large amplitude pressure oscillation in the Motor 1 test and the second test of Motor 2. The averaged chamber pressures of Motor 1 and Motor 2 lay between 542 and 625 psia, far less than the designed average pressure of 750 psia at LOX flow rate of 600 lbm/sec, as given in Table 5. In Motor 1 test and the second test of Motor 2, severe chamber pressure fluctuations (spikes) were noticed throughout the tests. Relatively small pressure peaks at the ending period are due to the onset of gaseous nitrogen for shutdown. In Motor 1 test 1 and Motor 2 test 2, each pressure spike using the high-speed data acquisition system (12500 data/sec) revealed similar characteristics of pressure build up and discharge processes. Magnitude of the spikes are generally close to the theoretical maximum operating pressure level while some surged as much as twice the mean pressure. Decrease in pressure timewise is expected, due to the throat erosion, lower flux level as the ports open up with subsequent lower fuel regression rates changing the O/F ratio.

C*

One of the ballistic parameters that quantifies motor performance is C*, a characteristic velocity shown in Figure 11. The ratio of actual C* to the theoretical maximum C* from the industry standard thermochemistry code represents motor efficiency. The C* efficiency in the figure indicates that a significant amount of fuel has not released all of its energy inside of the motor as previously experienced[28], as shown in Figure 25. Also, the C* efficiency seems to be higher in the motors with motor with higher O/F ratios. This phenomena has been observed in single port subscale motors[29]. Possible causes in the 250K-lb_f hybrid may be than the same mixing in the aft end of a motor may cause more combustion in an oxidizer rich environment or that the lower flux levels provided more reaction time in the ports and mixing chamber.

Regression rate

Direct measurement of the port circumferences were attempted using both mechanical (Crawler) and laser measuring devices to calculate the amount of fuel regressed. Figure 21 shows a typical pre-fire quad port configuration and Figures 22-24 show the average fuel regression rate of individual ports of Motor 2 acquired by the laser device. Notice that the regression profiles of the quad ports are not coincident with the result from the Crawler.[26] Also, note that direct impingement of oxidizer flow increases regression rate at the port as shown in Figure 22, Figure 23, and Figure 24.

In general in Motor 2, the regression rate increases monotonically lengthwise except the third test where the rate for the quad ports stay relatively in constant. From this result, it is obvious to consider dependency of the LOX flux level, motor length and port diameter in a fuel regression correlation.

Stability

A hybrid motor differs fundamentally in terms of combustion behavior compared with solid and liquid rockets in that the O/F ratio has an axial dependency. Historically, both acoustic and non-acoustic instabilities related to the motor geometry were encountered during the development of a large scale hybrid rocket motor. It is believed that the relatively cold flow of oxidizer in the head-end causes pressure oscillation and thus methods of adequate LOX vaporization, reduced droplet size, and use of flow deflector were introduced to suppress combustion instabilities.[28] Significant amount of efforts were given to evaluate combustion instability during the hybrid motor development in terms of vortex shedding[19] and diffusion flame movement[30], but complexity of the multi phase diffusion flame combustion dynamics in the turbulent reacting flow has not been fully disclosed yet.

From the instability point of view, it is not clear from Figure 22, Figure 23, and Figure 24 that, excluding deviation in the quad ports, significant difference in the regression rate at the center port from those at the quad ports in the second test leads to unstable combustion, since Motor 2 tests 1 and 3 were stable.

Variation of local O/F ratio in the combustion chambers may be a key factor for determining the hybrid motor stability. And multi-port with a one head-end LOX injector configuration having pressure variation between combustion ports from uneven distribution of oxidizer could be an additional source for the instability by developing pressure oscillation in tangential mode at the port entrance. The center ports in Motor 1 and Motor 2 are examples for the uneven distribution cases. Apparent local O/F ratio in the center port of Motor 1 seems lower than the optimum value from the entrance leading fuel rich condition through the entire port length. It could be a result of the flow deflector. In contrast to Motor 1, the center port of motor 2 has much higher oxidizer level at the inlet, allowing continuous increase of the fuel regression rate downstream with ongoing advantage of higher temperature. Even an excessive amount of LOX at inlet might cause the port entrance to be under two phase combustion.

AMROC apparently experienced the same situation with different regression rates in the CP and outer port and designed around it, since their large scale motors used only quad ports and blocked the center port. The blocked center port also acted as a splashblock, which increased the residence time in the forward chamber.[7,18]

A possible cause for pressure fluctuation in these motors is the difference of the fuel regression rate between upstream and downstream in the chamber and/or continuous throat erosion. A traveling wave in the combustion chamber disturbs turbulent mean flow field characteristics in the ports, which enhances mixing of unburned fuel and oxidizer in a periodic fashion and fuel regression rate by enhanced heat transfer to the fuel. From this point, gas filling and discharging sequence is being unbalanced until the chamber pressure reaches the maximum operating status. Continuous fuel regression and throat erosion disrupts the continuity by discharging more gases resulting in lowering the chamber pressure. This single port combustion phenomenon, along with the interaction with the other ports in a multiport design, could have lead to the instability caused in Motor 1 Test 1 and Motor 2 Test 2.

Nozzle

It was obvious that the reaction of carbon in the throat with hot oxygen in the exhausting gases accelerated the throat erosion. Real time erosion rate of the throat is not available because only pre- and post measurement were conducted. The results showed a higher erosion rate than predicted from the early subscale test data, with low O/F ratios[31]. However, later subscale tests with the same material indicated a similar erosion rate[16]. Different characteristics of the gas flow from the individual combustion ports caused irregular throat erosion aligned with the ports. This has been seen before in tests with multiport grains[31] and was expected.

Figure 33 shows thrust versus chamber pressure ratio for Motor 2 as an indirect indication of the throat erosion characteristics. The slope of the curve in the figure correlates with nozzle erosion rate. Note that discontinuities in the second test are due to the pressure peaks where instantaneous changes of the thrust coefficient, a dependent parameter on chamber pressure, occurred. Ignoring the discontinuities, throat erosion rate remains relatively constant, excluding the transient period of initial heating and charring at the beginning stages.

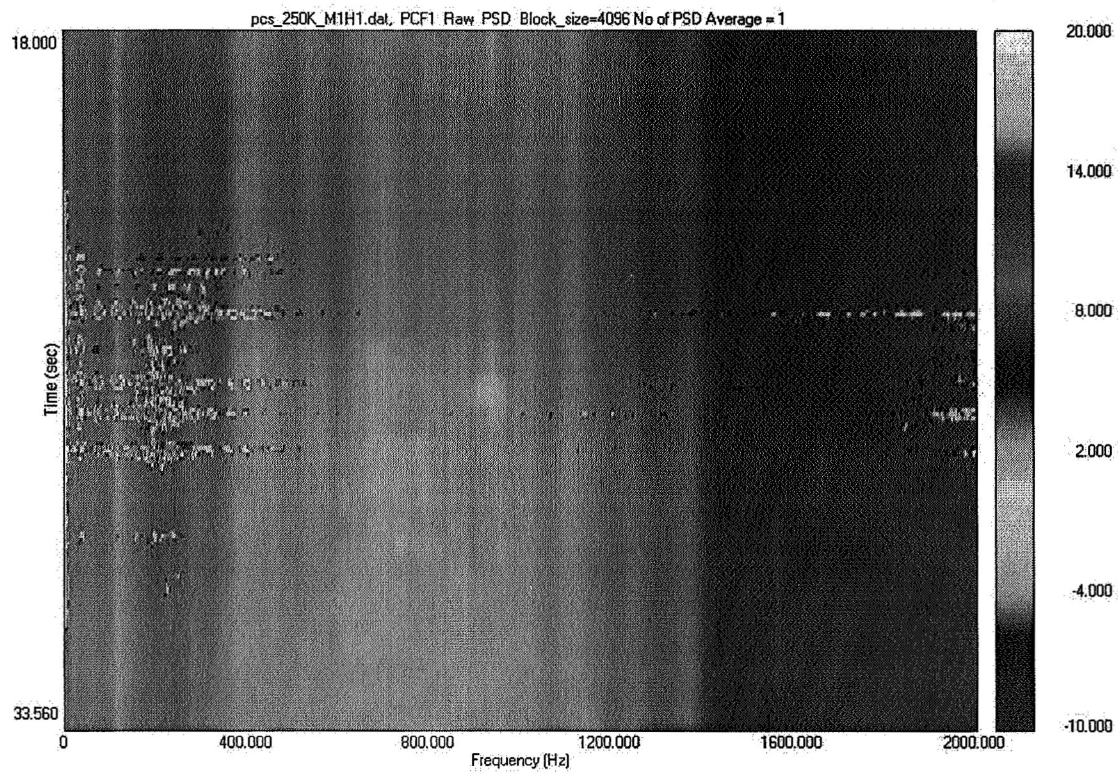


Figure 26 Motor 1 Test 1 Fwd PC Spectrogram[3]

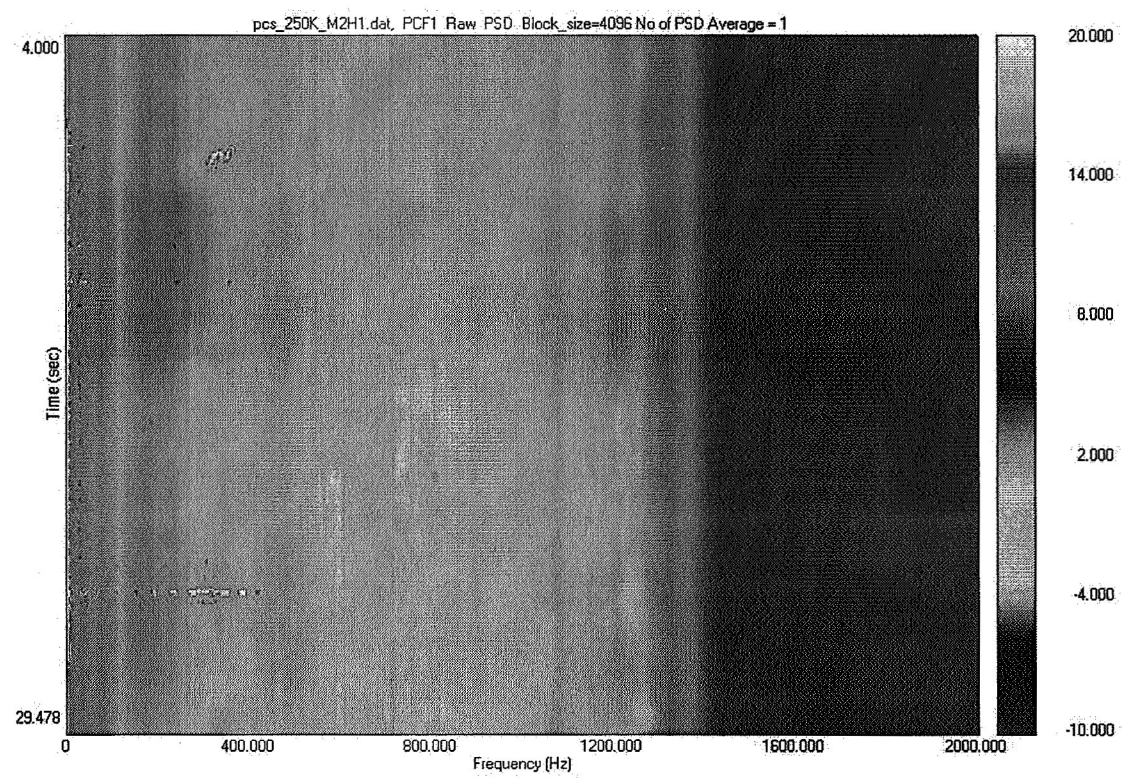


Figure 27 Motor 2 Test 1 Fwd PC Spectrogram[3]

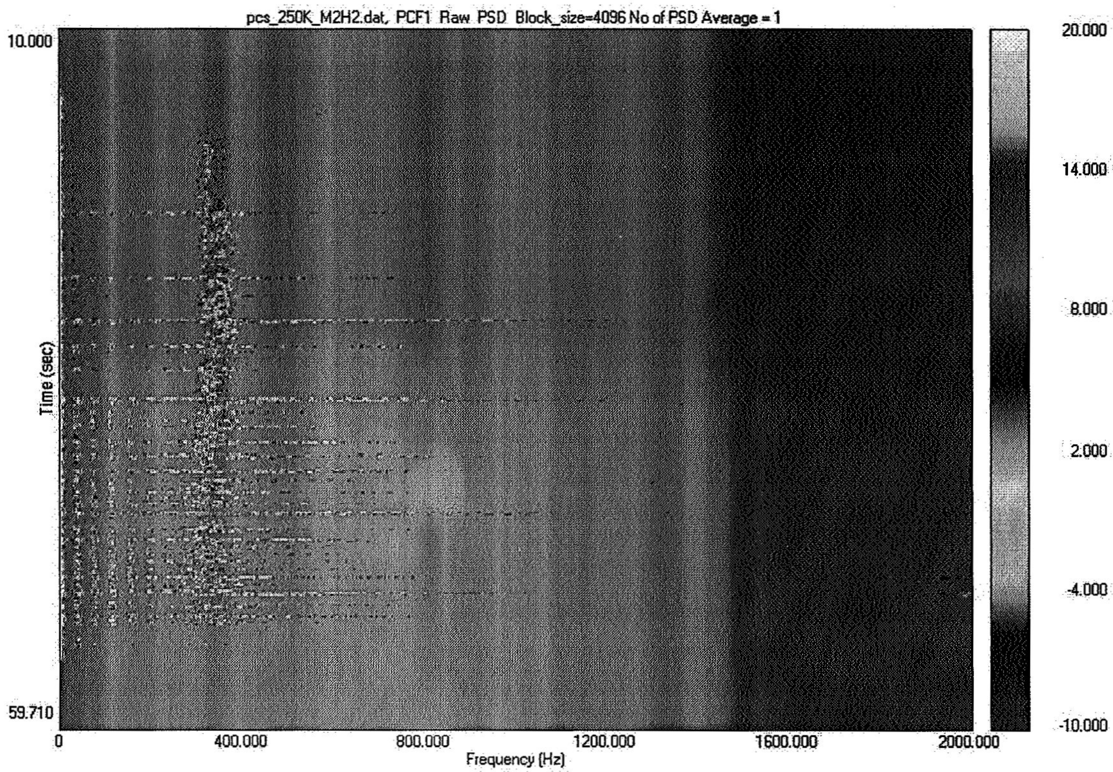


Figure 28 Motor 2 Test 2 Fwd PC Spectrogram[3]

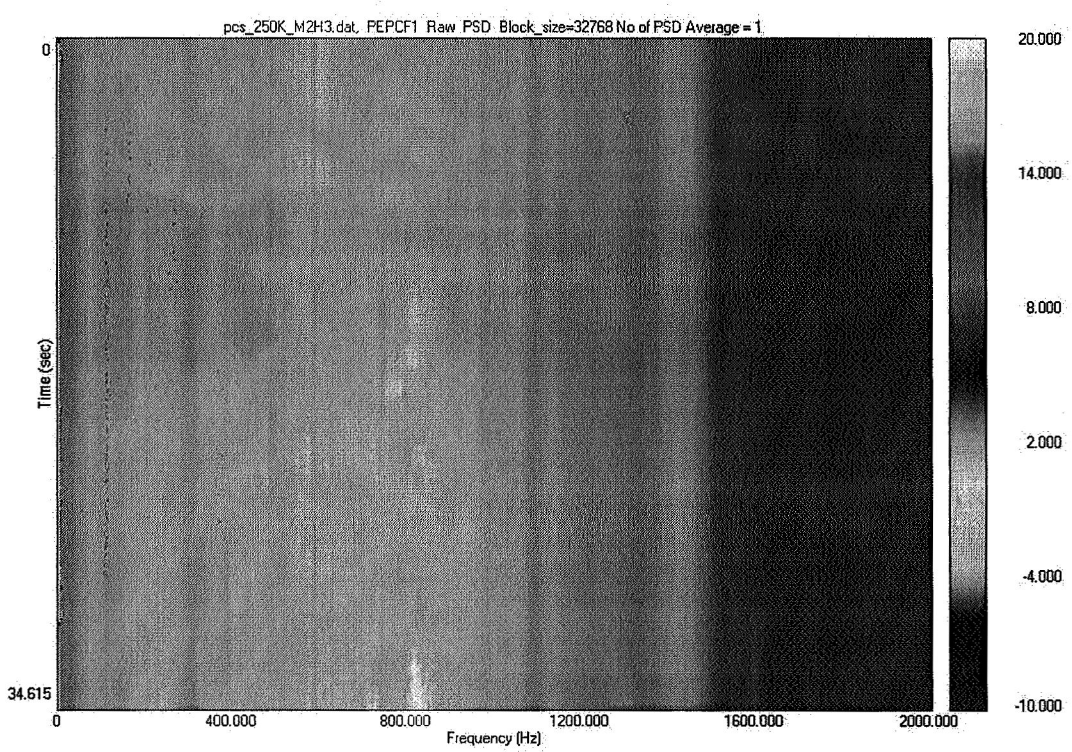


Figure 29 Motor 2 Test 3 Fwd PC Spectrogram[3]

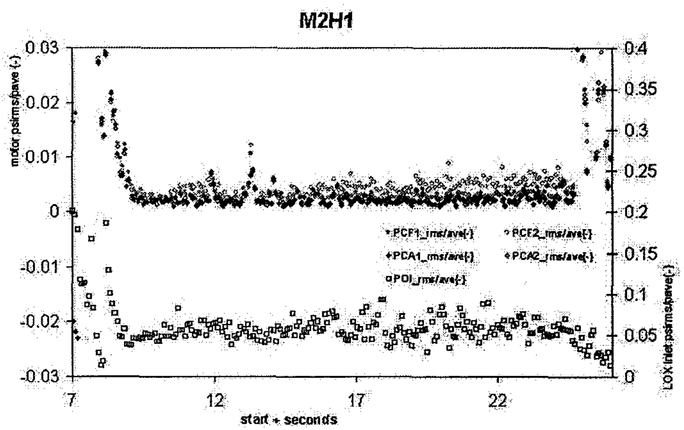


Figure 30 Filtered composites normalized by average pressure Motor 2 Test 1[3]

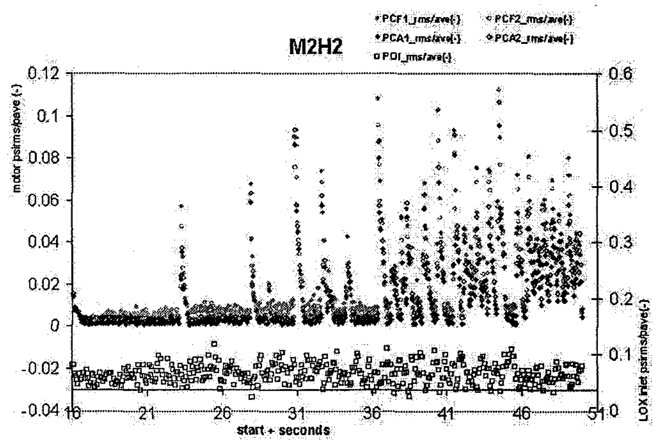


Figure 31 Filtered composites normalized by average pressure Motor 2 Test 2[3]

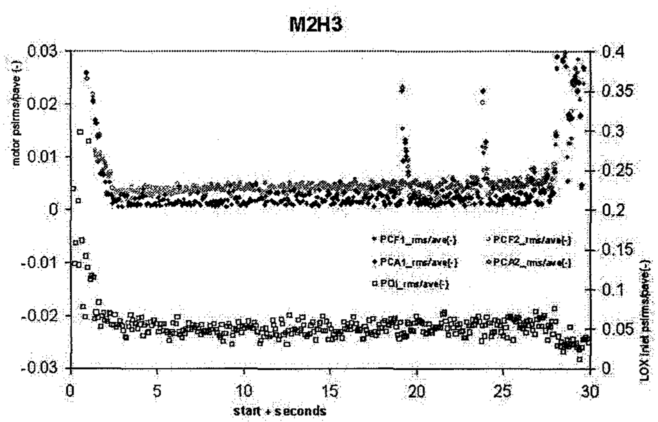


Figure 32 Filtered composites normalized by average pressure Motor 2 Test 3[3]

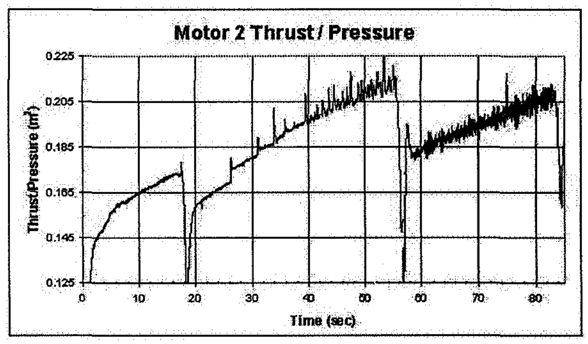


Figure 33 Motor 2 Tests Thrust/Pressure[3]

HPDP 250K-lb_f Conclusions

Motor 1's passive design was unstable. This doesn't imply that all hybrids of this size will require an active heat source in the front end of a hybrid, but this one was unsuccessful in achieving stable performance. Motor 2 was stable during tests 1 and 3, but drastically unstable in test 2. The concept to add heat in the head-end of the motor worked, but the design solution tested could not provide stability for the full 80 second duration. Another design solution will have to be worked for future full duration testing.

Scale up from small hybrids to large hybrids, as demonstrated by the achieved regression rates and lower than expected chamber pressures, was not done effectively on this program. Scale ups should be made from the largest port data possible.

The nozzle material selected for this program eroded greater than the design parameters.

HPDP 250K-lb_f testing, in some fashion, should continue. The ISP and stability observed in these tests provide an incentive to further improve this simple rocket system. Motor 1's grain has been fired for only 8 seconds and there have been several suggestions put forward for additional testing.

Scaled Composites SpaceShipOne

In 2004, a small hybrid rocket powered, privately developed spaceship was released from an airplane and flew to the edge of space. This vehicle was developed to win the X-Prize, and eventually had to be flown twice within two weeks. While the size of the SpaceShipOne's hybrid is not necessarily large in terms of thrust, its test setup and integration into the vehicle was shown to be novel and ingenious. In a test-what-you-fly mantra, Scaled Composites built a portable test stand that contained the flight oxidizer tank as well as the hybrid motor, in the in-flight configuration (See Figure 34). This allowed the propellant weights to be weighed directly on test stand, using sensors built into the device. This was especially necessary due to the use of Nitrous Oxide as an oxidizer. Nitrous Oxide properties are not as well documented as that of Liquid Oxygen or even hydrogen peroxide and the weight of the oxidizer tank and contents is a good way to keep track of the flow rates.

Figure 34 Scaled Composite Test Trailer[32]

Several items of interest include:

- the motor case being directly tied into the oxidizer tank, with the plumbing inside the tank(Figure 35). This reduces the need for additional structure to carry the load from the motor thru the tank. Also, it reduces the expansion/contraction issues with oxidizer loading – even though that should be minimal with room temperature nitrous oxide.
- The motor case and nozzle over wrap is a one piece item, with less leak paths due to less connection points.
- The hybrid had a cut off system in case of hybrid motor burn thru. A fiber optic wire was wrapped around the motor case and if it was broken/burnt thru during the flight, the oxidizer valve was to be commanded shut and the thrust terminated, increasing the safety of the system.
- It was also reported that in order to prove the insulation system was tolerant to the fuel burning irregularly, one motor case was fired for 2 times the normal 80 second burn time, with no burn thrus.

The basic sizing and design of the oxidizer tank and hybrid motor configuration was designed by Tim Pickens working directly for Scaled Composites. The specifics of the interior configuration (# of ports, fuel type, and ignition system) of the hybrid motor was designed and proposed by two competing companies with experience in hybrid rocket motor testing and development: Environmental Aerosciences Corporation (EAC) and Space Dev. Space Dev. had previously purchased the rights to AMROC technologies, patents and test data. EAC had been a participant in the development and flight of the HPDP N2O/HTPB sounding rockets launched out of NASA Wallops in 1996 and 1997. The internal configurations of the motors were slightly different, EAC proposed a single port design and SpaceDev proposed a 4 port design[33]. After a series of ground tests on the test trailer, Scaled Composites awarded the contract to Space Dev, and their motors have been flown on all the SpaceShipOne flights leading to winning the X-prize.

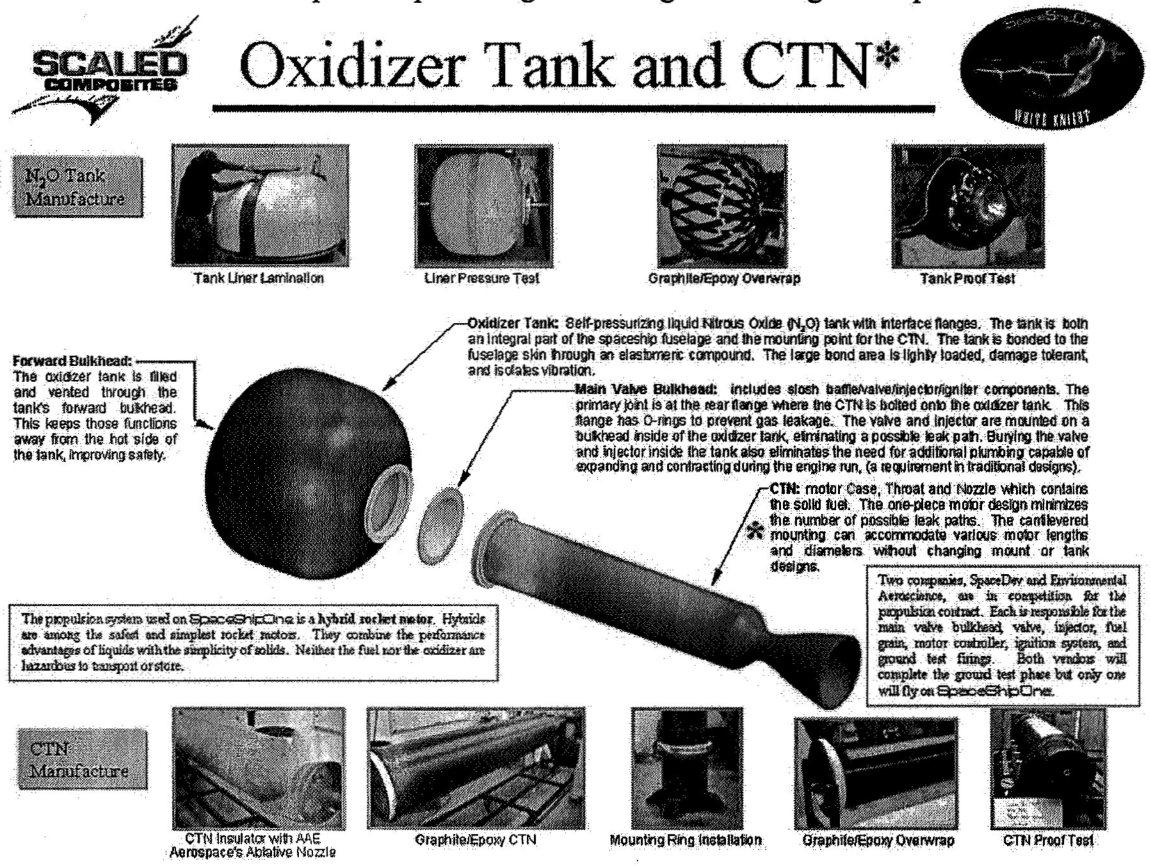


Figure 35 Scaled Composites Hybrid Motor[32]

Lockheed Martin/Darpa Falcon Testing

A large hybrid rocket motor was successfully test-fired Jan. 21, 2005 on the Air Force Research Laboratory's Test Stand 2-A on the ridge overlooking Edwards' dry lake bed and surrounding California's Mojave Desert. The test ran for the planned 60-second duration, and an initial review of data indicates that test objectives were met.

A second version of the motor was fired for 120 seconds on June 10, 2005. The second fuel grain was designed such that the 120-second test firing represented over 170 seconds of run time for the flight configuration.

The hybrid motors that were tested are full-scale versions of an upperstage motor and measures 11 feet in length and five feet in diameter. Besides the thrust size of this motor, ~23,500 pounds of thrust, and long duration, another item of interest about this testing is that the test was the first of a kind to fire a multi-port, multi-row hybrid motor.[34] While the details of this configuration have not been publicly released, the change in configuration might represent a step forward in the technology from the documented previous large scale hybrid testing.

These tests support the Defense Advanced Research Projects Agency (DARPA)/Air Force/NASA Falcon program, which is a 36-month long Phase II effort to develop and demonstrate an affordable and responsive space lift launcher capable of placing a small satellite, weighing 1,000 pounds, into a circular orbit of 100 nautical miles[35].

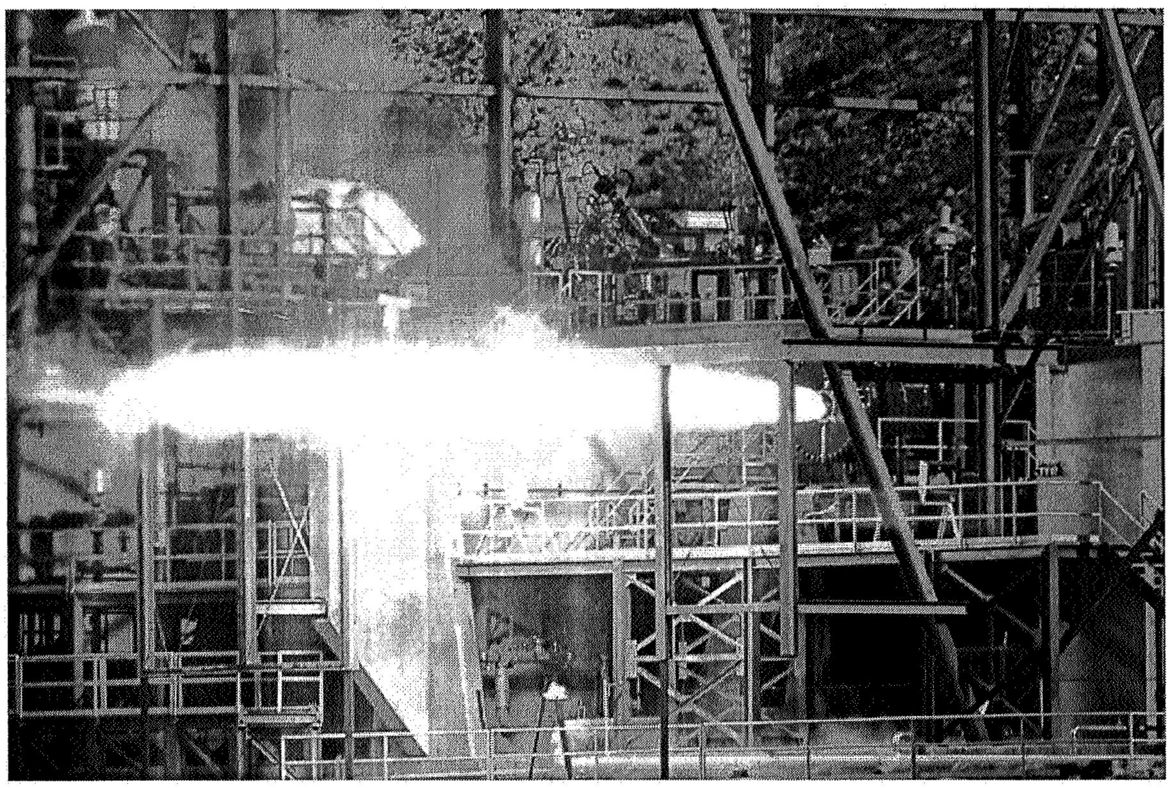


Figure 36 Lockheed Martin Falcon Test Jan. 21, 2005

Lockheed Martin's Planned Fuel Expulsion

One of the potential shortcomings of hybrid rockets is the residual fuel left remaining in a motor after the burn and the potential failure modes of a multi-port web breaking off and damaging the nozzle or plugging the throat. Previous design solutions have been shown to increase the fuel strength by web stiffeners,[36] over build the web thickness to eliminate the concern by intentionally leaving the residual web overly thick (works well for ballistic motors and HPDP 250K-lbf), or just over design the system to account for the residual fuel. The problem with these solutions is it doesn't optimize the hybrid for a flight configuration, since the motor has to accelerate that inert residual fuel mass. An optimal solution would be the fuel remain in place and continue burning until it was wafer thin.

Lockheed Martin has investigated the fuel expulsion issue and investigated at what point would the fuel fail, and how best to deal with it. A structural model was developed based on a beam model, using pressure differences between the ports as the loading of the beam. Since the pressure in individual ports is difficult to model, an approximation was made that the difference between the forward and aft chamber pressure could be the limiting case for the port to port differential pressure. That fuel structural model was integrated into a ballistics code, where once a piece of fuel got to where it would analytically fail and would break off, it no longer contributed to the ballistic performance of the motor. They performed two tests of a 10-inch diameter hybrid motor – one with a low tensile strength fuel and another with a higher tensile strength fuel. The low tensile strength fuel failed at a residual web thickness of approximately one inch and the effect was visible in the pressure trace. The high tensile strength fuel lasted much longer in the burn and started breaking at a web thickness of approximately 0.155 inches. Due to the noise in the pressure trace, it's difficult to determine when the fuel broke loose, but the implications of the testing are clear: a high tensile fuel may permit the web thickness to remain intact in the motor longer and break off only when the parts are small - allowing hybrid rockets to burn to almost depletion on the fuel side, increasing the system performance by lowering the inert weight and lowering the risk of the potential fuel failure modes since the fuel segments are so small.[37]

Conclusions

Hybrid motors have been demonstrated at large sizes with good success. These successes and the inherent safety and simplicity of hybrids have lead hybrids to be the propulsion system of choice for a privately developed manned application.

AMROC's experience with the DM-01 demonstrates that a large scale hybrid can be designed, fabricated and tested in 13 months. SpaceShipOne's hybrid motor was also developed, integrated with the flight system and flown in just a few years. Manufacturing processes to develop and build hybrid rocket motors are available and ready to use.

Scale up from small hybrids to large hybrids, as demonstrated by the achieved regression rates and lower than expected chamber pressures, was not done effectively on the HPDP 250K-lb_f program. Burn rates, derived from multi-port motors with port hydraulic diameters of 2, were used to design the HPDP motor, without any adjustment for port size. The designed HPDP 250K-lb_f port diameters were on the order of 4 times as large as the motor from which the burnrate came from, without any verification of those burnrates at near the final size until much too late in the program. Inaccurate predictions of regression rate on the large scale, based on subscale data, show up in higher than planned oxidizer to fuel ratios and result in lower chamber pressure and thrust values. The HPDP 250K-lb_f missed the regression rates by ~30% and therefore missed the target pressures and thrust values. However, on AMROC's large motors, the scale up was done effectively – small scale burn rates were tested on larger full sized bi-port motors, which allowed the burnrates to be adjusted before the final design of the full scale multiport motor. For conventional hybrids with forward oxidizer injection, scale ups should be made from the largest port data possible. Testing of hybrids should be as close to the flight size as possible to understand the possible lower regression rates and stability issues that appear in larger motors.

AMROC lead the way in solving the non-acoustic instability (NAI) combustion stability concerns associated with large scale hybrid rockets. They successfully demonstrated a solution with the stable DM-01 and DM-02 firings. HPDP demonstrated that the similar combustion stability could be obtained with heat addition from the forward end provided by smaller hybrid motors.

Most large scale hybrids have been tested with pressure fed systems. Pressure fed systems have been employed successfully on the hybrid flight systems (Firebolt, SpaceShipOne). Larger flight systems may need to be powered by pump fed systems. AMROC, AlliedSignal Aerospace and NASA's Stennis Space Center have demonstrated that hybrids can have stable combustion with a pump fed LOX system at a meaningful thrust and flow level.

There is still work that needs to be completed in large scale hybrids to make hybrids more competitive. The oxidizer-rich hybrid combustion products have shown to have higher than expected erosion rates on some of the traditional solid rocket motor throat materials. Lower throat erosion would lead to an increase in the performance of the motors and therefore is an area that needs to be investigated further. Another concern that needs to be addressed further is lowering the residual burnout mass of hybrid motors – a solution has been demonstrated at a representative scale, with high tensile fuel that remains in place until web is very thin, that shows promise, but also needs to be demonstrated on large multi-port grains. Analytical tools need to be developed to predict the fuel regression rates as the combustion ports are scaled up in size - currently this must be done empirically.

Hybrid Motors at the 250 K-lb_f size have been demonstrated they can be stable at or near the design thrusts and pressures by both HPDP and AMROC testing. Subsequent designs should reap the lessons learned from those programs and lead to a stable and efficient hybrid rocket motor design for large scale applications.

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