Genetic Algorithm Optimization of a Cost Competitive Hybrid Rocket Booster

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Performance, reliability and cost have always been drivers in the rocket business. Hybrid rockets have been late entries into the launch business due to substantial early development work on liquid rockets and solid rockets. Slowly the technology readiness level of hybrids has been increasing due to various large scale testing and flight tests of hybrid rockets. One remaining issue is the cost of hybrids vs the existing launch propulsion systems. This paper will review the known state-of-the-art hybrid development work to date and incorporate it into a genetic algorithm to optimize the configuration based on various parameters. A cost module will be incorporated to the code based on the weights of the components. The design will be optimized on meeting the performance requirements at the lowest cost.

I. Introduction

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 design simplicity, on-demand operational characteristics and 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.

Hybrid Rockets have been used in only one flight-production application (Teledyne Ryan AQM-81A 'Firebolt Supersonic Aerial Target) and one series of recent manned flight demonstrations (Burt Rutan's SpaceshipOne), suggesting that advantages have been overlooked in some potential applications. Hybrids are soon to fly on Virgin Galactic's SpaceShipTwo and were on Sierra Nevada's Dream Chaser vehicle initially. These 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 – Most 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.

Operations - Due to the nature of the combustion (and in most cases lack of solid additives), the fuel grains are very robust, cracks are inconsequential. During operation, the lack of premixing between the solid fuel and oxidizer eliminates that as a possible detonate able mixture. Since the fuel regresses due to vaporization from the flame front, there is little fuel grain temperature sensitivity to the regression rates.

Casting – When compared to Solid rocket motors, hybrids are safer to manufacture, assemble and transport due to inert grains and non-explosive solid fuel ingredients. Classical hybrid motors can be cast in light industrial facilities using the techniques used in traditional solid propellant casting. Even though hybrids are insensitive to cracks and defects in the propellant, gross disturbances in the flow

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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. However, hybrids use only one fluid system, which make them less complex than bi-liquid systems (liquid rocket engines). Compared to liquids, hybrids have half the plumbing system with simplified throttling, shut down, and steady state operations, since only the liquid flow rate is controlled.

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. Thrust termination is simply accomplished by turning off the liquid flow rate.

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 Polybutadiene based fuel - LOX rocket is equivalent to a RP-1-LOX liquid engine, and significantly higher than a solid rocket motor, ~40 seconds higher than a Polybutadiene/AL/AP system. Other fuel and oxidizer combinations yield higher and lower performance values, with different system issues to work with.

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. However, quantification of the cost is difficult to prove.

Hybrid Rocket development has suffered due to some potential disadvantages. The nature of the combustion produces a much lower regression rate than a solid rocket propellant. That low regression rate means multiple ports are required for the same thrust or a technique/different fuel system needs to be found for a higher regression rate. These multiport ports can yield a hybrid system that has a low bulk density or volumetric fuel loading. There have been observed cases of fuel ejection during motor operation and the corners in the ports can lead to residual propellant slivers. Due to the boundary layer mixing, there can be low combustion efficiency due to diffusion flames and poor mixing. As the hybrid motor burns, the fuel flow rate changes over time, which can result in an O/F shift. Most of these disadvantages can be overcome with design solutions.

One of the remaining issues with Hybrid Rocket motors is financial. Is it cost competitive to use a hybrid in a launch system?

A top level study by Matthias Grosse in 2007, "Design Challenges for a Cost competitive Hybrid Rocket Booster"[27], indicated that a hybrid rocket booster was more expensive than an equivalent solid rocket booster or a liquid rocket booster. That analysis was done using a single point design extrapolated to a much larger size with various weight ratio estimates from solid and liquid systems without optimizing the hybrid system based on cost. This paper documents an attempt optimize a booster design based on cost, using the cost indices of functional units from that 2007 study.

II. Past Hybrid Booster Activities

There have been several successful and not so successful hybrid rocket efforts in the past. These have been documented in several places[27,1]. For this paper, we'll review concepts that are relevant to the design envelope being discussed here.

A. AMROC 250K-lb$_f$ Motor Development

After the failed launch of SET-1[2] 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.[3] **Figure 1** shows the grain design, which was also a one row design, with a blocked off center port.

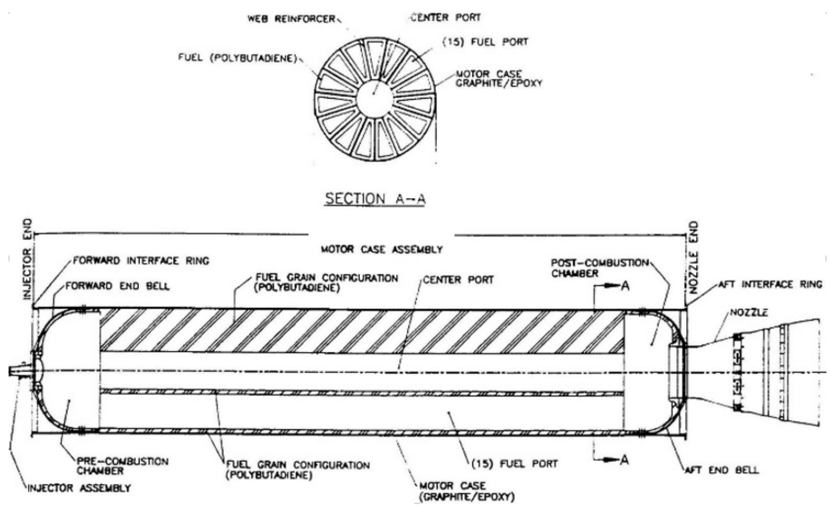

Figure 1 AMROC DM-01 250K-lbf Hybrid Motor[3]

Table 1 H-250K-lbf Design Parameters[3]

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. The total project duration, from initial design to the completion of testing, was thirteen months.[3]

Test results

The DM-01 motor was tested in a series of four static firings. 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.[3] The motors were relatively stable and performance data is shown in **Table 2**.

Table 2 AMROC 250K-lb_f DM-01 Test Results[3]

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 Press (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[4]. However, the motor was only fired once due to technical and funding issues.

AMROC 250K-lb_f Conclusions related to this effort

The AMROC work demonstrated that a hybrid could be developed quickly and produce a stable motor. A scale up of the configuration is presented.

B. 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.[5,6] 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.

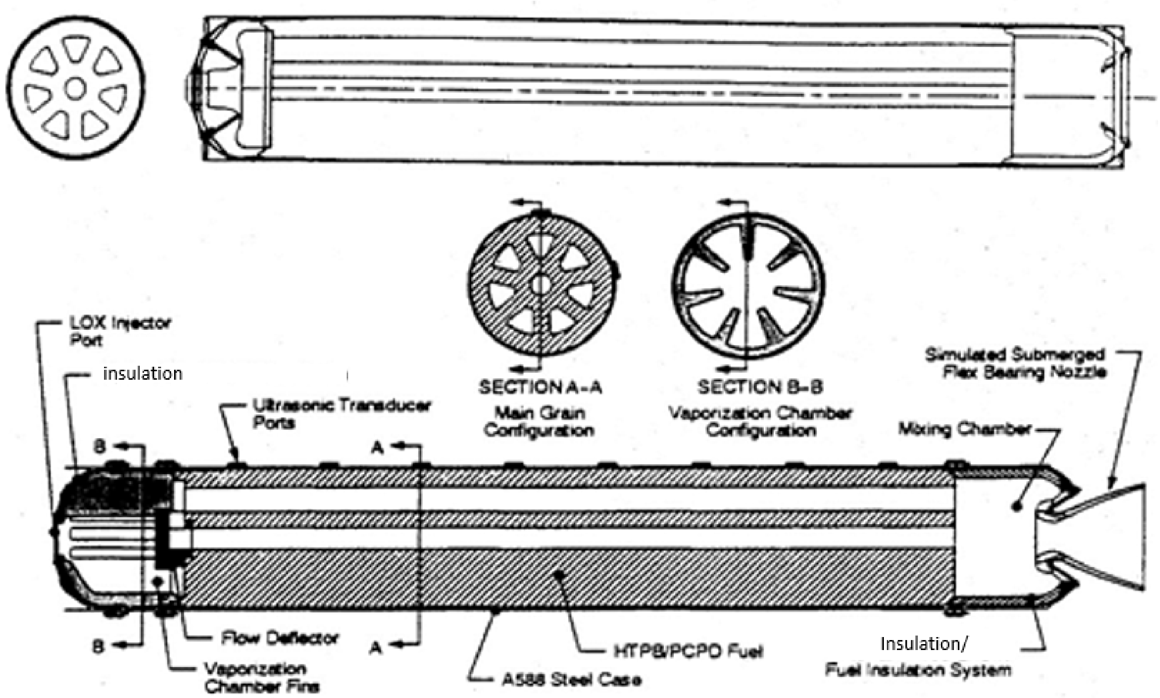

Figure 2 - 250K-lb$_f$ HPDP Hybrid Motor Layout[7]

11-inch and 24-inch Motor testing at MSFC

There had 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.[8,9,10,11] 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 3. Details of the injector, fuel grain, nozzle design are given in references 5 and 6. A photo of a pretest aft end of the grain is shown in Figure 3. Note that this design also included one row of ports, but with a center burning port. The design was built for ground testing with no hints as to flight weight components. At the end of the scheduled full burn time, the motor was to have roughly 2 inches of web fuel left between the ports.

Figure 3 250K-lb$_f$ HPDP Motor Ports[7]

Table 3 250K-lb$_f$ Design Parameters[7]

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	Polybutadiene*/LOX

* Polybutadiene 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,[12,13,14] 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).

Combustion stability and head end designs are addressed in more detail in reference 42. In summary, it was found that active heat addition in the forward end of the motor was necessary for the stable combustions of Lox/Polybutadiene motors of this size. The active heat addition technique used here was small heater motors located in the head end of the forward dome.

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. Ignition was smooth and combustion was stable (Figure 4). 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.

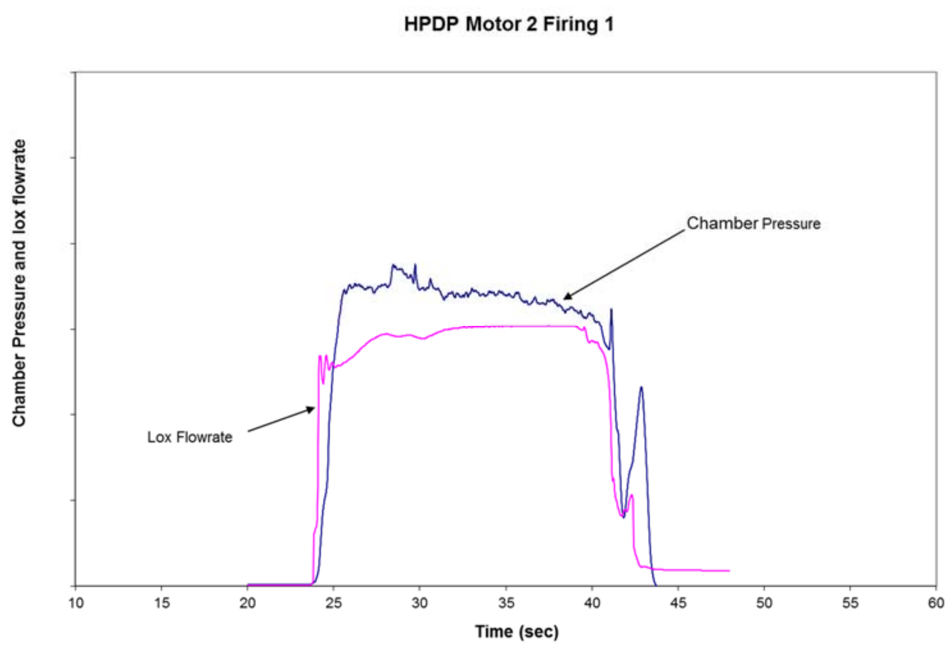

Figure 4 HPDP 250K-lb$_f$ Motor 2 Firing 1[7]

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 5). 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.

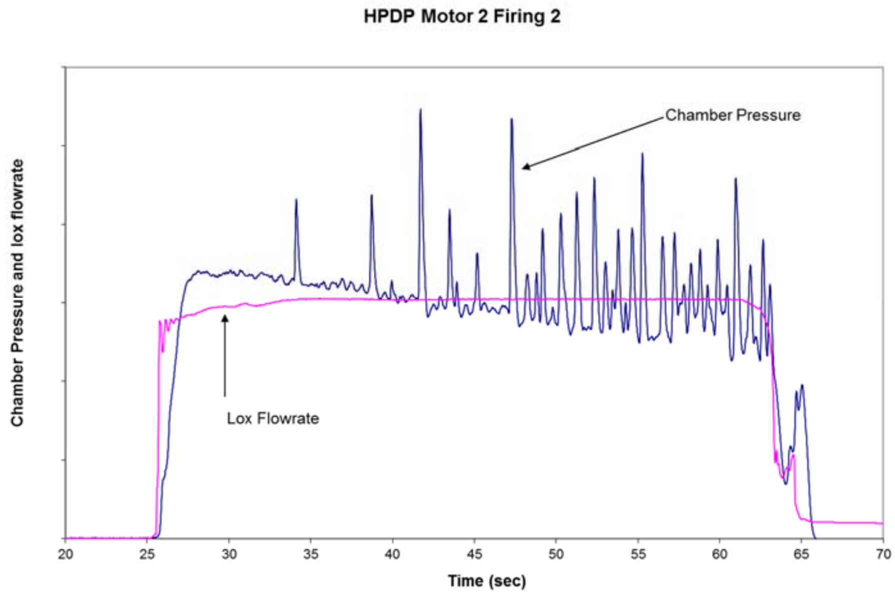

Figure 5 HPDP 250K-lb$_f$ Motor 2 Firing 2[7]

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 6). 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.

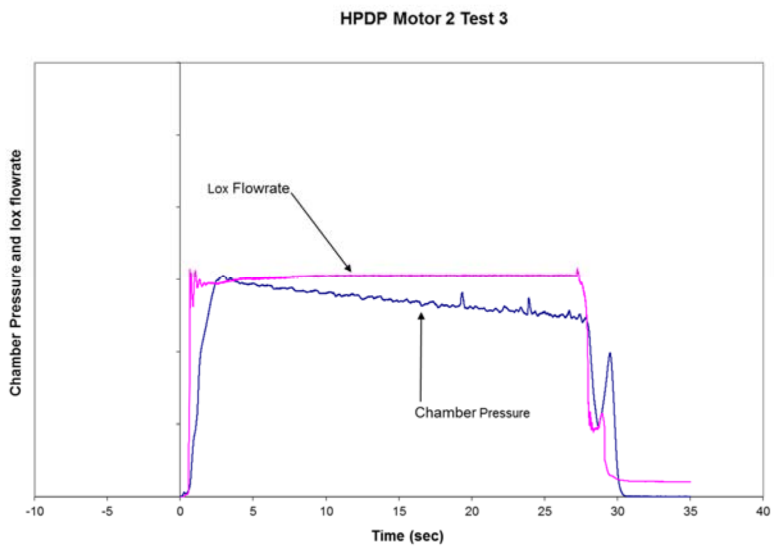

Figure 6 HPDP 250K-lb$_f$ Motor 2 Firing 3[7]

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[15]. 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 7.

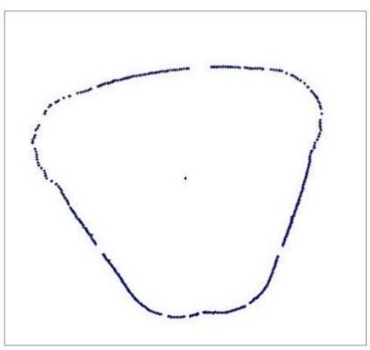

Figure 7 Laser Port Mapping Sample – Pretest Port[7]

Average regression rate data the ports per test can be shown in Figure 8, Figure 9 and Figure 10. There was a significant difference between the three weighing techniques, with the maximum percentage differences of techniques near 10%. This has led 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.

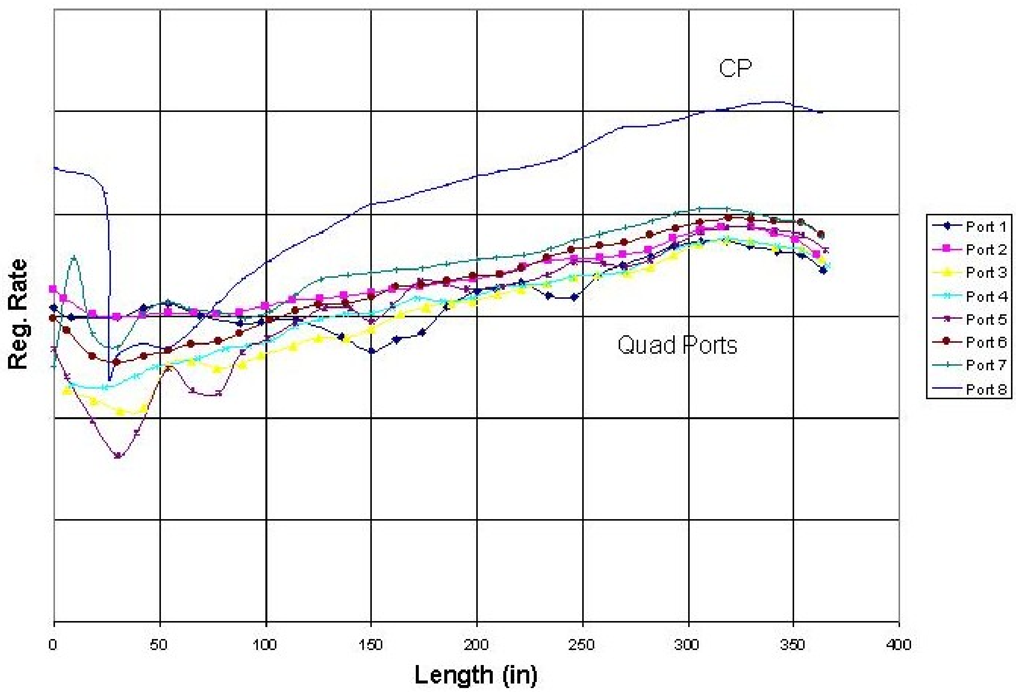

Figure 8 HPDP 250K-lb$_f$ Motor 2 Test 1 Regression Rates(Center Port and Quad Ports)[7]

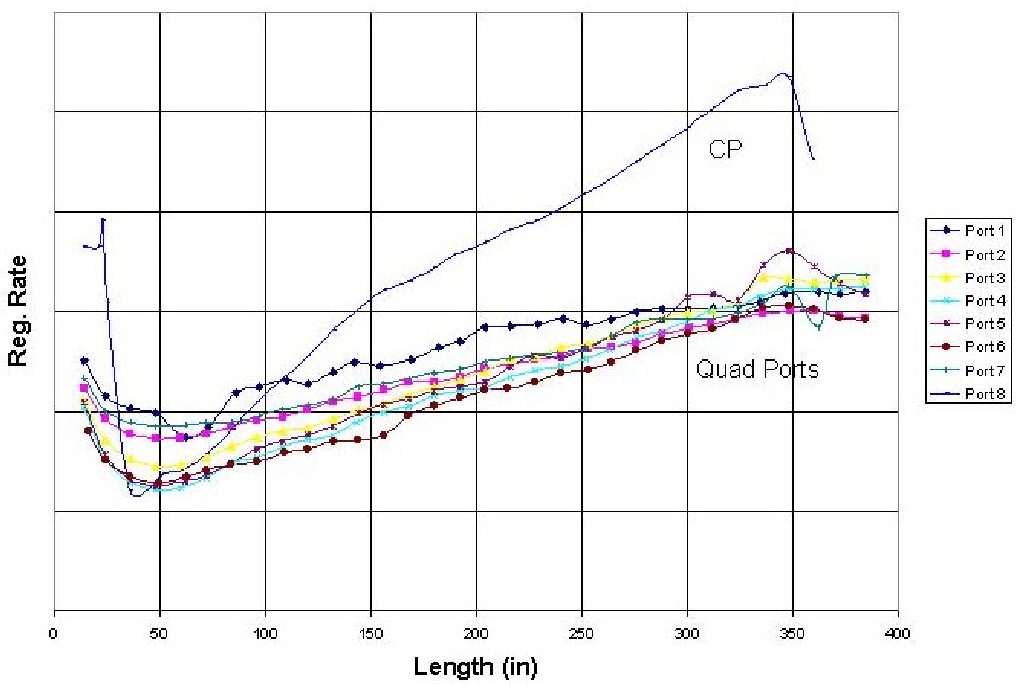

Figure 9 HPDP 250K-lb$_f$ Motor 2 Test 2 Regression Rates(Center Port and Quad Ports)[7]

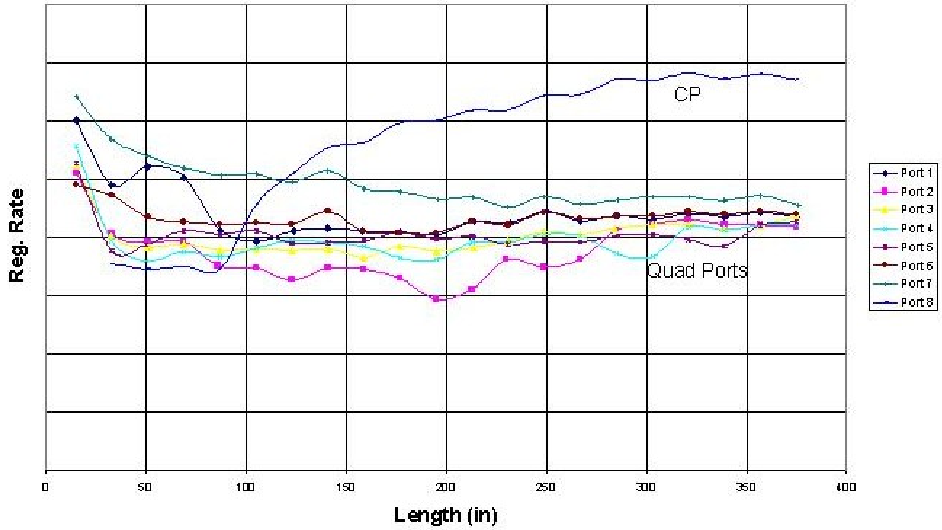

Figure 10 HPDP 250K-lbf Motor 2 Test 3 Regression Rates(Center Port and Quad Ports)[7]

The 250K-lbf motor regression rate plots show that, as a general trend, the aft end of the motor port burns out faster than the forward end. There are some entrance areas where there is higher regression. That can also be seen in forward and aft end pictures in **Figure 11** and **Figure 12**.

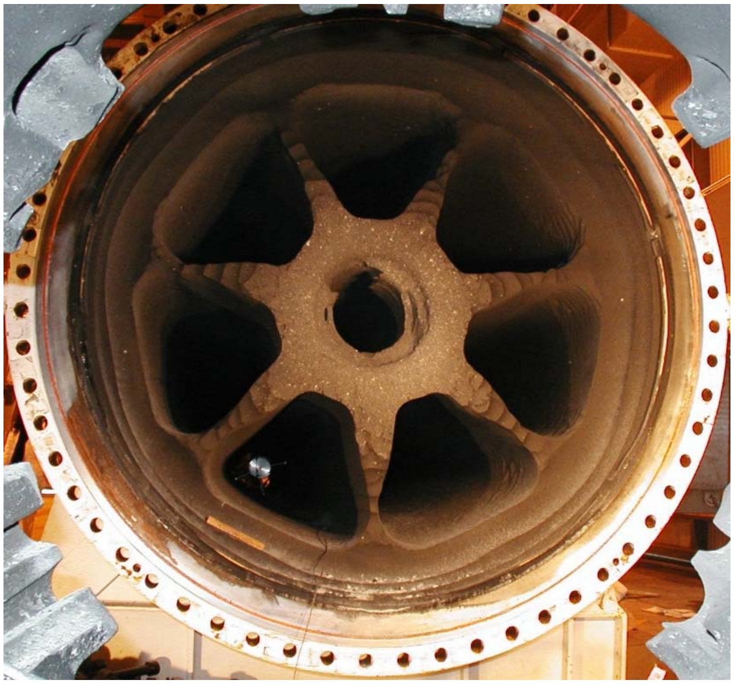

Figure 11 HPDP Motor 2 Fwd End Post Test 3

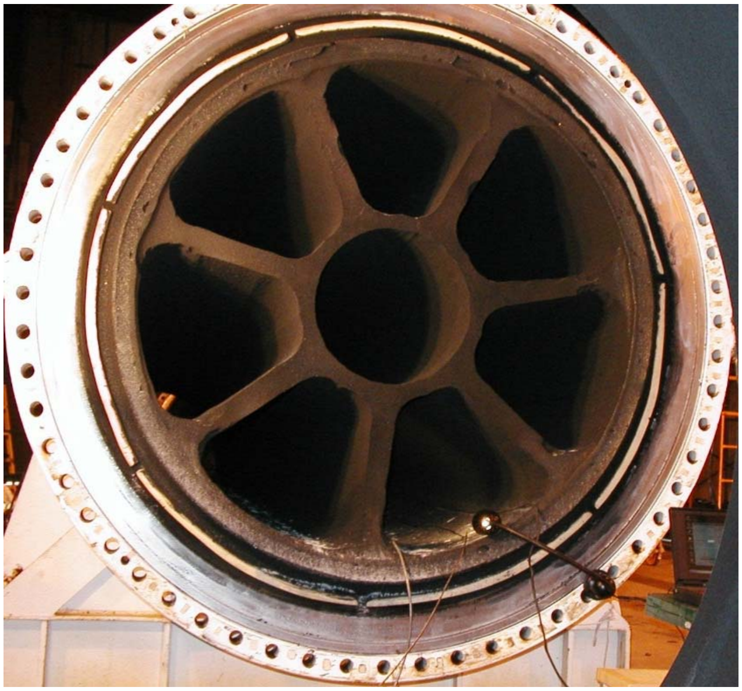

Figure 12 HPDP Motor 2 Aft End Post Test 3

Table 2. Average Motor Performance Parameters[7]

Parameter	Motor 1	Motor 2		
	Test 1	Test 1	Test 2	Test 3
Thrust (lb_f) - Vac	177,137	186,337	210,065	195,989
ISP VAC	250.0	248.6	276.9	263.9
ISP VAC EFF	0.77	0.78	0.90	0.92
Cstar	4,576	4,856	5,093	5,044
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 13. The test O/F and ISP/CSTAR calculations are from HPDP final report[16] with Laser mapping of center port weights.

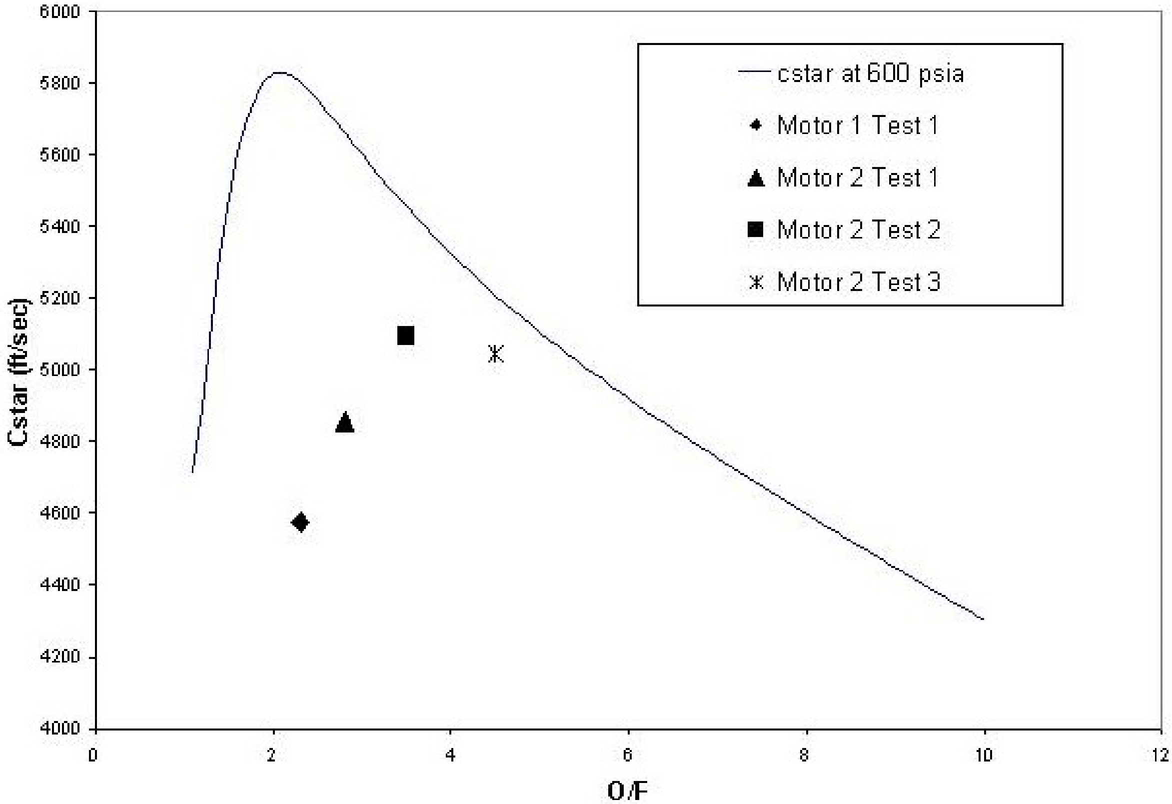

Figure 13 Theoretical Cstar vs Data[7]

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 the regression rates of a motor with a hydraulic port diameter of 2 inches. 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[43]. Other work comparing small ports regression rates extrapolated to larger ports showed an error in the regression rates greater than 10%[17]. Reduction in the fuel flow rate affects the O/F, chamber pressure and thrust.

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[15]. 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 shown in Figure 4, Figure 5, and Figure 6 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 3. 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 13 Theoretical Cstar vs Data[7]**. 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[18], as shown in Figure 13.

Also, the C* efficiency seems to be higher in the motors with higher O/F ratios. This phenomena has been observed in single port subscale motors[19]. 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 7 shows a typical pre-fire quad port configuration and **Figure 8** thru **Figure 10** 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.[15] Also, note that direct impingement of oxidizer flow increases regression rate at the port as shown in Figure 8, Figure 9, and Figure 10.

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.

HPDP 250K-lb_f Conclusions related to this effort

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. A concept to add heat to the head end of the motor to ensure stability is included in the analysis presented in this paper.

In general, the aft end regresses faster than the forward end of the motor. The phenomena of the aft end of the ports regressing faster than the forward end of the ports is taken advantage of in this paper.

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. In this analysis, regression rates are based on the best data available and a function has been included for effects of larger or smaller ports.

C. Lockheed Martin Hybrid Sounding Rocket (HYSR)

A large-scale hybrid rocket was successfully launched from the NASA Wallops Flight Facility on December 18, 2002 as a technology demonstration for hybrid propulsion and related subsystems. The Hybrid Sounding Rocket (HYSR) program started in 1999. The overall goal of the program was to develop a single stage propulsion system capable of replacing existing two and three stage sounding rockets, with additional objectives to demonstrate the required technology for the launch of a large scale hybrid sounding rocket booster, to demonstrate the positive attributes of hybrid propulsion, to demonstrate two hybrid-based subsystems, and to advance the Technology Readiness Level of hybrid propulsion. The hybrid rocket had a propellant combination of liquid oxygen and polybutadiene based fuel and produced approximately 60,000 lb of vacuum thrust. The three year technology demonstration program was a collaborative effort between NASA and Lockheed Martin and had a total budget under $6M, which was a combination of Lockheed Martin and NASA funding. The program advanced Technology Readiness Level and performance of hybrid propulsion.[20]

Heavy Weight Motor Testing took place from February to September 2000 at Stennis Space Center's E-3 Complex. Four full-scale motors were tested to gain fuel regression, stability, and performance data to validate the fuel grain design for the HYSR. The fuel that was cast into the motor cases was a mixture of Polybutadiene and high percentage of aluminum to optimize the delivered energy of the system. The motor case center segment had 5 penetrations equally spaced around the circumference of the motor with weldolets used for pass-through Swagelok ™ connectors. These penetrations were used to route the gaseous oxygen to the Lockheed Martin patented staged combustion system (U.S. Patent 5,794,435), which is used for ignition and maintaining combustion stability throughout flight. Figure 14 shows a sketch of the motor case used for Task A testing.

Figure 14 HYSR Heavy weight motor case used for Task A testing[20]

There were two center segment fuel grain configurations tested in Task A due to a redesign that was necessary after Test 2. The initial fuel grain design incorporated 10 quadrilateral ports around a single circular center port. After the first two tests, it was determined that the initial design flux (total mass flow rate divided by the port cross section area) was too aggressive and the port size was increased for future tests to correct this problem.

Figure 15 is a plot of the vacuum thrust from Test 4 of the heavyweight motor test series. The motor was tested for approximately 20 seconds with a planned throttle at 8 seconds into the burn. The vacuum thrust is regressive versus time due to the reduction in propellant flow rate versus time and, to a lesser effect, nozzle erosion. The increase in thrust during the ignition transient was caused by solid fuel being ejected from the motor. The fuel failed due to high loading and weak fuel tensile strength. After the HySR program, Lockheed Martin investigated how fuel failures occur and developed a possible solution to that problem for future hybrids.[21]

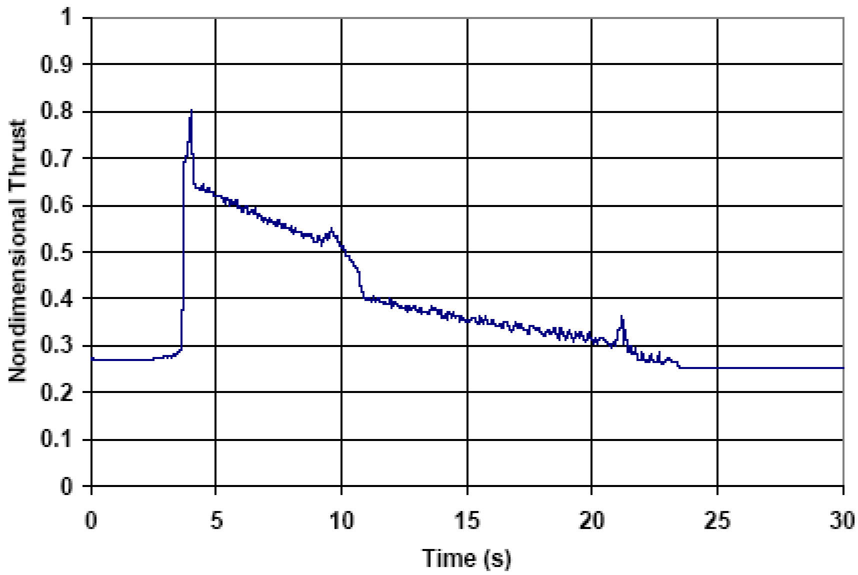

Figure 15 Vacuum thrust versus time for Test 4[20]

Other than the initial ignition transient, the motor performed within a ± 2.5 % stability band around the mean value. The motor also demonstrated a high C* and ISP efficiency.

Flight Activities

On the final launch attempt, the LOX and helium tanks were loaded in under 60 minutes. The LOX tank was pre-pressurized to 745 psig at T-10 seconds and the ignition sequence was commanded. After 3 years of development, the HYSR was launched from the Wallops Flight Facility on December 18, 2002 at 6:15 AM E.S.T.

After an analysis of the flight data, it was concluded that the vehicle achieved an altitude of approximately 42 km, impacted 65 km downrange, and had a time of flight of 213 seconds. The initial acceleration of the vehicle from the rail was approximately 6.1 g's, which was determined from high speed video of the vehicle on the launch rail. The burn time of the motor was approximately 33.4 seconds and the timed despin and payload ejection events occurred as planned. Although the performance was lower than predicted, the factors that reduced the altitude could be explained and performance could be recovered in future missions.

Lockheed Martin HYSR Conclusions related to this effort

HYSR program developed and flew a concept for keeping heat in the head end of the motor. While that particular configuration is not used in this analysis, it lead to the concept being used in the analysis.

Also, the HYSR program developed an aluminum loaded Polybutadiene fuel combination that was not able to handle the loads of the high flux being used and the high accelerations encountered during rocket liftoff. This resulted in unburned fuel that was ejected from the motor during ignition and ascent. This lead to Lockheed Martin researching Polybutadiene combinations and processes to get to a higher capability fuels.

D. Lockheed Martin/Darpa Falcon Testing

Lockheed Martin conducted risk reduction testing to 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[22].

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.

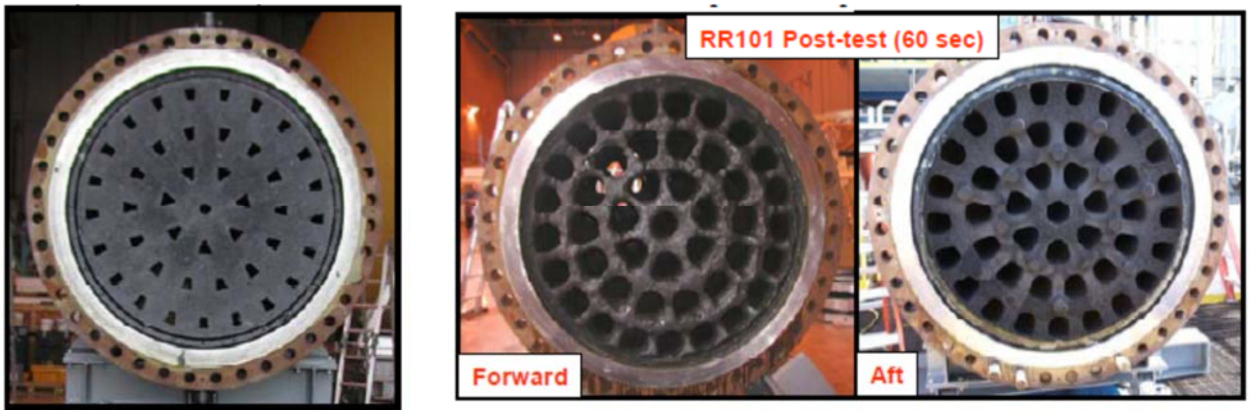

Figure 16 Lockheed Martin RR101 Pre and Post Test Photos - 3 Row -43 Port Fuel Grain[29]

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 (see **Figure 17**).

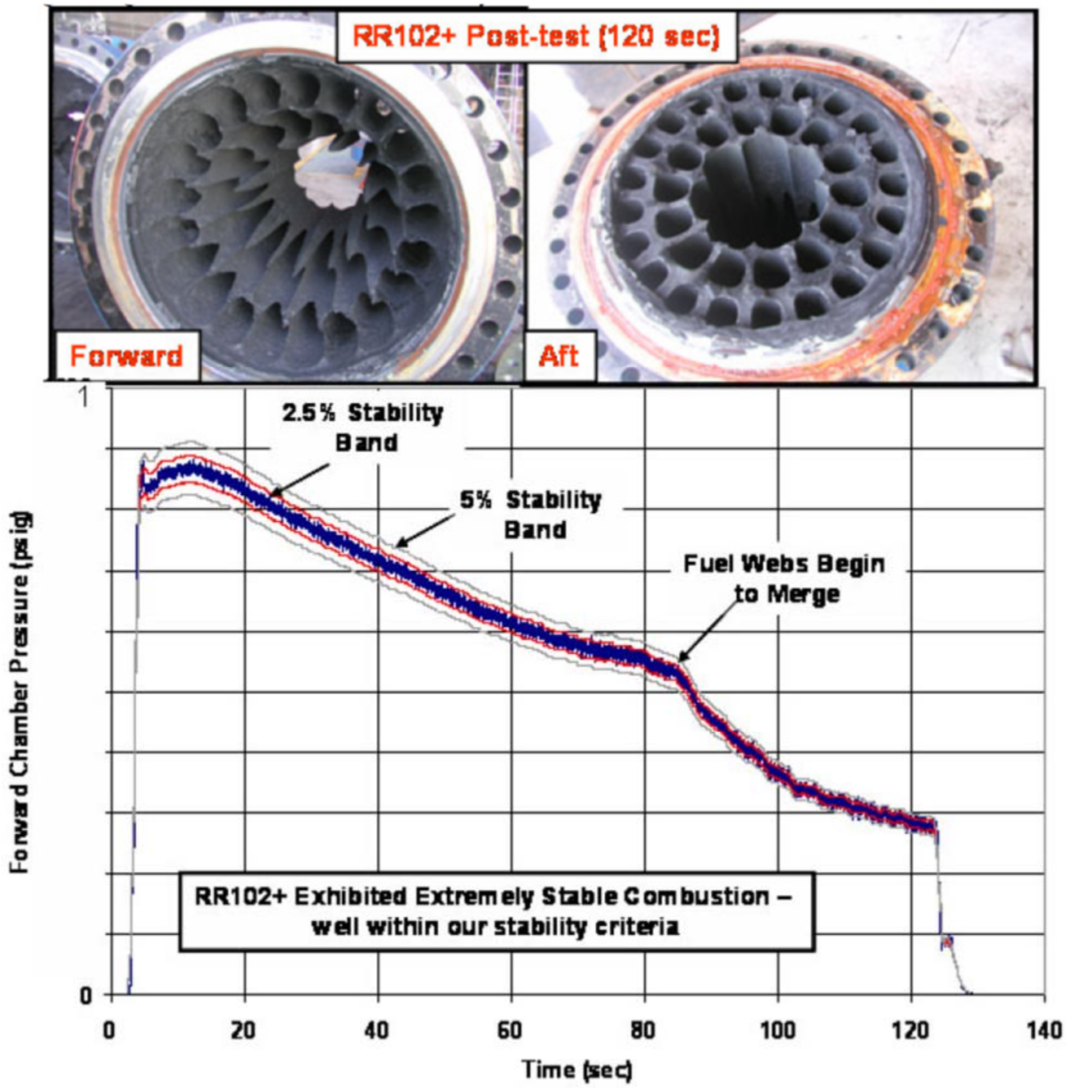

Figure 17 RR102+ Post Test Photos and Chamber Pressure[29]

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 to fire a multi-port, multi-row hybrid motor.[23]

The use of multi-port, multi-row grains allow helps correct one of the design problems of hybrid motors, the tendency to design long, thin motors. The use of multi-port, multi-row grains allows for shorter, squatter boosters.

In order to get multiport, multi-row grains to work, another potential shortcoming of hybrid rockets had to be addressed, 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. In some cases, the fuel is ejected on the pad [HYSR] or lost during firing [Spaceship One and HPDP/AMROC 250k]. Previous design solutions have been shown to increase the fuel strength by web stiffeners,[24] 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.

Prior to the Darpa Falcon risk reduction testing, Lockheed Martin had investigated the fuel residual/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.[25] Lockheed Martin's solution was to develop a fuel formulation where the tensile strength was greatly increased (Figure 18).

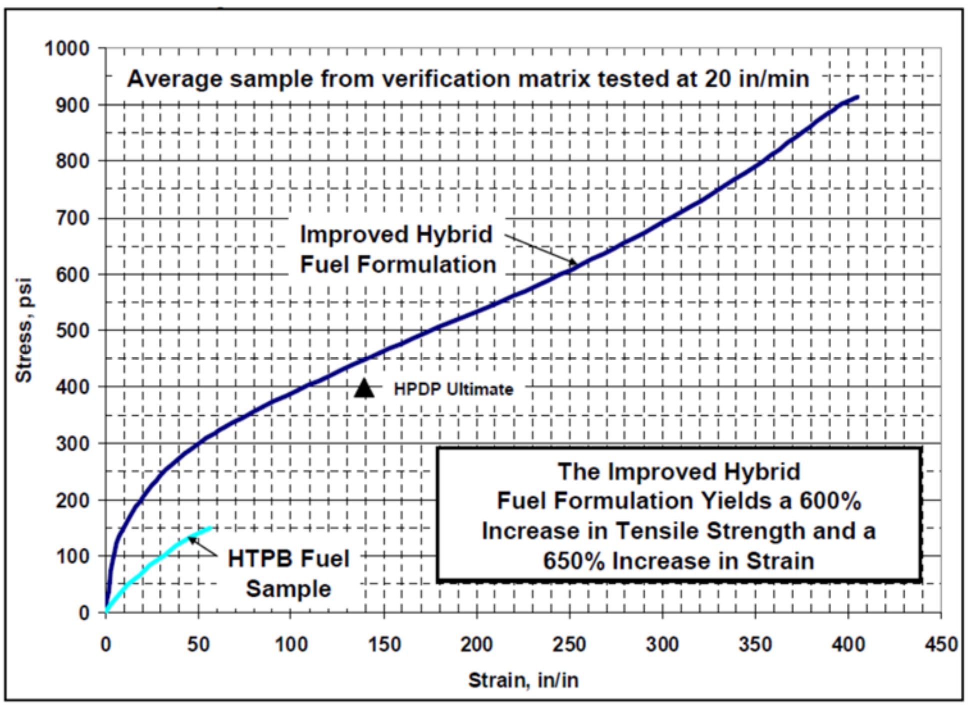

Figure 18 Tensile Strength and Strain of LMF-900 Fuel vs. Previous Fuels[29]

That allows the webs to burn down to a thin web section and when it does fail the parts are small enough that it doesn't affect the chamber pressure. That fuel formulation was demonstrated in the hardware shown in Figure 17, with the slivers burned or ejected from the motor with no effect on the chamber pressure. Previously, that had been demonstrated in the NASA/Industry Joint Internal Research and Development (JIRAD) program on an 11 inch motor.

Figure 19 JIRAD Ports Showing Thin Web is an example of a grain burned back to a thin section.

Figure 19 JIRAD Ports Showing Thin Web

Lockheed Martin/Darpa Falcon Conclusions related to this effort

Lockheed Martin designed their lox injector injector/grain configuration to burn out from the inside out, that is the center port to inner ports burn out first, keeping the outer support grains still structurally there as shown in **Figure 17**. The concept has merit, but there is a potential consequence of an unmixed oxidizer core going thru the motor late in burn. For the modeling effort of this paper, the grain will be burned from the aft end forward. The strategy is that the web will remain intact over time in the forward end of the motor and the fuel/oxidizer mixing will be better since there is not an open core of the motor.

Another concept included in this paper from the Lockheed Martin Darpa work[29] include canned heater motors. The concept is to remove the heat addition from inside the motor and move it outside the motor chamber and lower the length of the motor case. A difference in this effort is that the heater motors are being sized as vortex hybrids.

Lockheed Martin's Darpa lox rich Gas Generator[30,29] was also used in this paper to drive a turbine for the lox turbo pump.

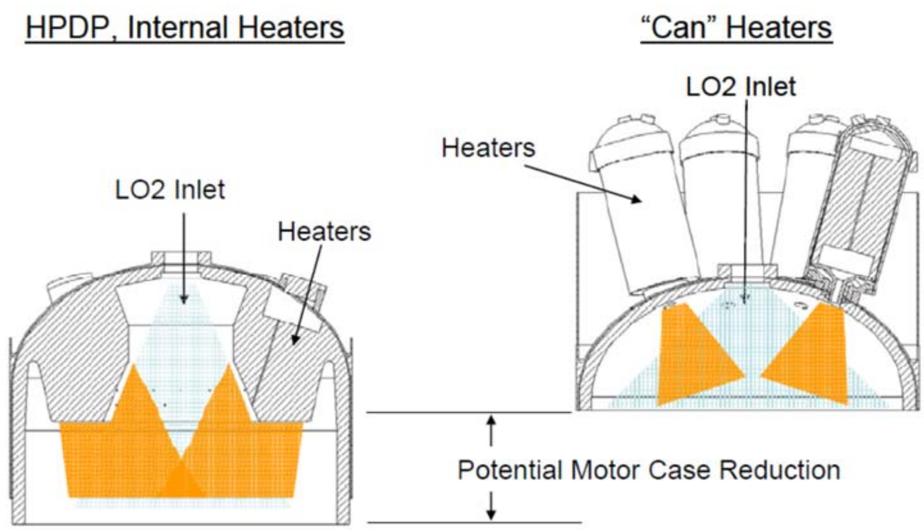

Figure 20 Lockheed Martin "Can" Heaters Concept[29]

III. Mission and Model details

A. Reference Booster and Mission

The reference mission and cost reference data is based on the comparison study by Grosse[27]. The Ariane 5 has a series of heavy launch vehicles that use a pair of solid strap ons, the Ariane 5 EAP. Currently there are 3 Ariane 5 vehicles variations that use the dual solid strap-ons, but this study will try to match previous work. That work did a cost comparison study between the baseline solids, a previous liquid rocket study and a hybrids solutions. Using cost models for the various components from the solid and liquid system, the hybrid system had a higher cost then the solid baseline and liquid booster alternative.

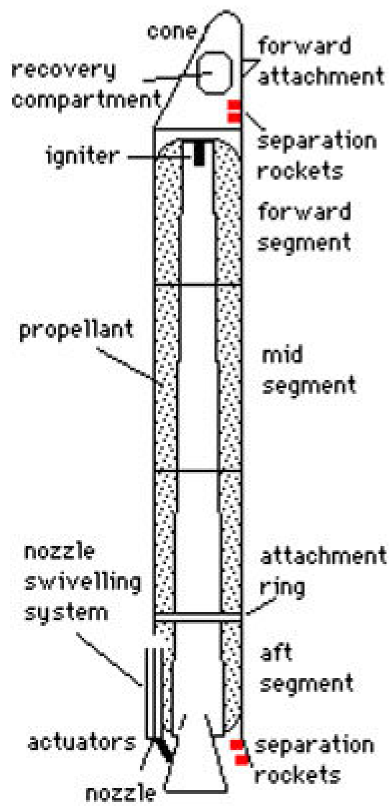

Figure 21Ariane Launcher Booster[26]

To make valid comparisons, this analysis is based on the same reference vehicle, a 240 t constant mass core. The boost describes a provided "the modeled parallel staged boosters are strapped on a core vehicle with a constant mass of 240 t. A core vehicle thrust is only considered for calculation of the required booster's liquid engine or solid rocket nozzle mass. The booster's mission is to produce an ideal velocity increment of 2.5 km/s based on its vacuum specific impulse. For simplification, Grosse used a constant propellant mass flow it is assumed for all boosters. According to the baseline data base of the H-1800 and 250-K based hybrid booster, the average vacuum thrust-to-initial weight ratio is set for all three types of boosters to 2.6 g, also if this is not optimal to maximize the launch vehicle's payload (resulting total vehicle initial acceleration is 1.6-1.8 g)."[27]

The solid Reference Booster is based on Ariane 5 EAP. The liquid Reference Counterpart is based on Astriums' EAL study + F-1 type engine to replace RD-180.

B. Configuration of the Hybrid Booster in this analysis

1) Range of inputs. These are the variables that were changed and evaluated as part of the optimization process.

   a. Fuel type – Fuel type defines the fuel and oxidizer type, Cstar look up tables and regression rates constants. For this analysis, looking at a LOX Polybutadiene combination and a LOX Polybutadiene with Aluminum loading in the fuel.

   b. Number of ports – This input reflects the number of ports in the first row of ports. The center port is assumed to be burning. This input, along with the number of rows, drives the configuration of the grain. The inputs are limited from 4 to 9 ports in the multirow configuration.

c. Number of rows – This affects the volumetric efficiency of a booster. Based on previous analysis in Ref 29, the change from 3 to 4 rows doesn't lessen the void space greatly, so the model was originally limited from 1 to 3 rows. It was later expanded to 7 rows.

d. Chamber Pressure – Initial Chamber pressure selection drives motor case thickness and turbo pump requirements. A wide range of chamber pressure inputs were used from 300 to 1300 psia.

e. Initial Flux – Initial port flux was used in sizing initial ports. The HPDP 250K hybrid motors have had fluxes of 0.64[28]. While a higher initial flux does lead to an initial higher fuel rate, it can also create a large change in the oxidizer to Fuel ratio during the burn. A higher flux port design may result in a longer booster than starting with a lower flux level, per an analysis in Ref 29. A range of initial fluxes from 0.4 to 1.0 are allowed.

f. Number of heater motors – Based on the concept of canned heater motors[29], trying to see if the number of heater motors would make a difference. Varied the number of heater motors from 8 to 22. Used a vortex type heater motor for simplicity; however did not do any post run fit checks to see if the motors would fit in the intertank region. The concept is for the LOX to run the vortex chamber to cool the throat as it gasifies the LOX.

g. Lox tank pressure - The lox tank pressure is a critical function in the sizing of the lox tank mass and for sizing the turbopump mass, due to head pressure requirements. Lox pressure was allowed to vary from 15 to 165 psi.

h. Lox ullage gas temperature – The mass of the ullage gas is defined by the tank size, pressure and temperature. The ullage gas mass is considered payload in this case and needs to be minimized.

i. Burn time – The rocket equation controls the amount of propellant required for a certain delta velocity, however that typically optimized for low thrust to weight motors. In order to get higher thrust to weight motors, the impulse had to be delivered over shorter periods of time. To get this to work, 'extra points' were given to the evaluated function when the thrust to weight was in the right range. The burn time was varied from 60 to 130 seconds, in 10 second steps.

j. Nozzle expansion – This was originally a variable, but after further review, but decided it would be more simple to have the nozzle expansion fixed to 9 psia. All these motors would fire at sea level and based on a rule of thumb that if the expansion was to 9 psia, there would not be any worry about flow separation in the nozzle, which would drive up the loads in the nozzle, and therefore the weight.

C. Details of the design

a. The hybrid motor grain – the hybrid motor is designed based on the fuel type, number of ports, number of rows, chamber pressure, initial flux, burn time and nozzle expansion as direct inputs. The number of heater motors is an indirect input since that is driven by the oxidizer flow rate, however the heater flow counts as mass flow into the forward dome, and is evaluated in the motor ballistics. The grain can have 1 to 7 rows of ports, depending on the input.

b. The forward and aft domes are fuel lined ¾ ellipses with a ½ inch layer of an insulator on the inside.

c. The lox injector is based on 2 X the wall thickness and equal to the diameter of the pipe upstream of the injector

d. The nozzle is sized based on Humble's[38] empirical nozzle sizing calculations in section 7.6.4.

e. The TVC weight is an approximation, assuming ½ the nozzle weight per Humble[38] section 6.3.8.

f. The motor case is based on a composite, with the outer diameter set by the hybrid grain outer web thickness, with a ½ inch layer of an insulator on the inside.

g. Connecting the lox tank to the turbo pump and then to the motor injector is a pipe/valve/venturi system. The line is sized based on the turbopump pressure (which is motor pressure * a factor), oxidizer flow and hybrid motor diameter. The weight of the pipe/valve/venturi system is estimated at 2X the weight of the pipe.

h. Turbopump is based on Humble's [38] liquid propulsion section 5.4.

i. A hybrid gas generator drives the turbo pump, so the turbine will see an oxidizer rich gas. Some testing of that system was done in references [30] and [29]. The hybrid motor is sized as a vortex motor.

j. The exhaust gas from the hybrid gas generator, after it goes thru the turbine, goes thru a heat exchanger to flash lox to gox for ullage pressurant. A line to carry the ullage gas to the tank is sized based on the tank ullage gas conditions. The weight of the pipe, valves and heat exchanger is approximated as 3X the weight of the pipe.

k. A vent valve/line for lox tank filling is sized for the top of the lox tank.

l. The lox tank is sized based on the required oxidizer flow for the motor, heaters, press system and gas generator and the lox tank pressures.

m. Heater motors are required for stable operation, by ensuring oxidizer vaporization in the forward dome. Heater motors are based off the 'Canned Stage Combustion System' concept described in reference [29], but with vortex motors.

n. The intertank and aft skirts are based on a representative length to cover the distance and support the weight.

o. Equipment weights booster separation motors were scaled from the Space Shuttle SRBs. The other equipment weights were taken from another program's estimates.

D. Other items of interest

a. Lockheed Martin's work indicated that a way to burn the motor grain out was to design the grain and injector to concentrate oxidizer down the middle of the grain, so the web burns from the inside out. When the fuel web was a minimum thickness, fuel could release with no damage. This was made possible by the use of high tensile strength fuel. In this code, the grain webs are iteratively adjusted so the aft end, that is all the rows of fuel webs, burns out at the same time. The hybrid grain burns out from the back toward the front. In this analysis, 'nsegchk' is an input variable indicating what grain segment is left intact when the motor stops burning. In these runs there are 10 segments in the calculations, so a nsegchk of 5 would indicate 5 of 10 segments/ half the grain would be remaining in the case, with the web sections aft of that burnt out. This value is set at the beginning of a genetic algorithm run.

b. Web slivers, that is the web in the corners of the ports thicker than the normal web, still burns aft the web between the ports is burnt. The burnrate is based on what the burnrate would be if the port were at its largest size before the web burns thru.

E. Genetic Algorithm

1) Background of code – “Very briefly, a genetic algorithm is a search/optimization technique based on natural selection. Successive generations evolve more fit individuals based on Darwinian survival of the fittest. The genetic algorithm is a computer simulation of such evolution where the user provides the environment (function) in which the population must evolve.”[31] The particular Genetic Algorithm code being used was downloaded from the web [reference 31] in the late 1990s after reading another paper on genetic algorithms used in the development of hybrid rocket motor designs.[32] Genetic algorithms have been used to size multiple rocket configurations.[33,34,35,36,37]

2) Summary of Code - The basic code flow was copied from Reference [32]. The genetic algorithm initially makes 50 sets of random zeros and ones. These sets represent the genes in the genetic algorithm. The genes are then interpreted as inputs by the hybrid code, where a few of the characteristics are, for instance, an initial chamber pressure, so these are the characteristics of the hybrid booster being evaluated. The inputs are fed into hybrid evaluation code similar to **Figure 22** to get an output function. The ‘better’ output function characteristics are kept, the lesser ones are discarded. The kept function characteristics are used to generate new pairs of random zeros and ones for the next generation. This is a survival of the fittest concept.

3) The code takes the input and sizes a hybrid booster. The code includes a hybrid ballistics model that runs every iteration and based on the burn out characteristics, updates the web thicknesses so the web thicknesses are equivalent and adjusts the length of the grain so the average O/F is close to the best for that oxidizer fuel combination. Included in the code are ‘design modules’ to estimates for the weights of the various components. Some of the ‘design modules’ are quite involved (the hybrid ballistics code), others are empirical estimates (nozzle weight is an empirically from Humble[38]) and others just rough estimates (TVC weight is ½ weight of nozzle, Humble 6.3.8). The code converges on the hybrid booster design when the difference in between the input and output of the burnout weight and ISP are within a certain tolerance. Decreasing the tolerance can greatly increase the precision and run time and for this exercise the tolerances was set at 5%. For a quick sensitivity analysis, the tolerances for the minimum cost parameters were run at different settings, see **Table 4**. As shown, difference in total cost and total dry weight due to convergence check tolerance level is numbers is small compared to the rough estimate of the analysis.

Table 4 Sensitivity Analysis of Convergence check tolerance level

Convergence check tolerance level	Total Cost	Total dry weight	# of iterations required to converge
5%	136019	427467	3
1%	140634	442834	6
0.1%	141142	444679	7

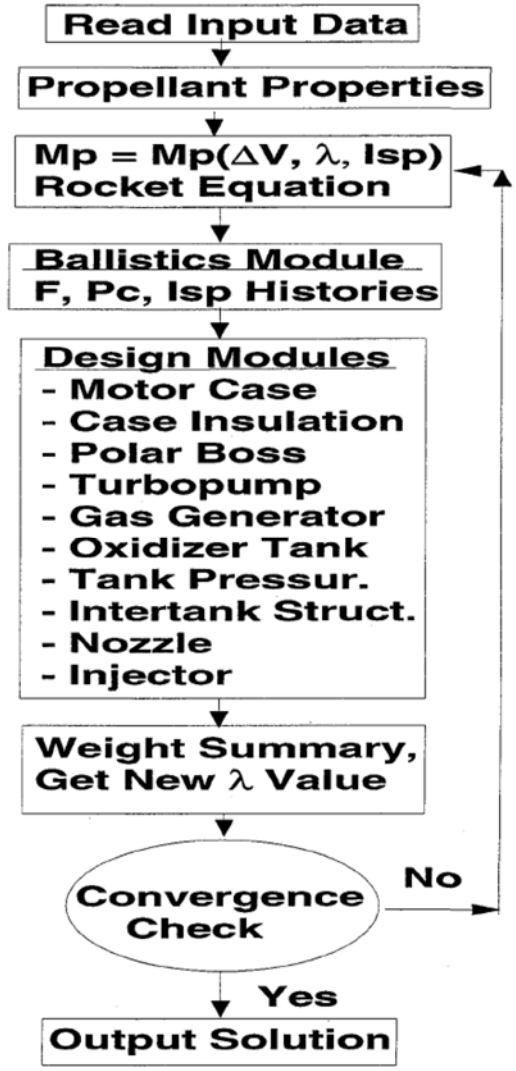

Figure 22 Flowchart for HYROCS code[32]

4) Performance numbers – The code includes all the 'Functional Units' listed below and most of the 'Related Components'. A review of AIAA S-120-2006 Standard Mass Properties Control for Space Systems[39] indicated that some level of Mass Growth Allowance(MGA) was required for this effort, so a uniform 20% MGA was included for all the 'related components' that were sized, which rolled up into 20% for all the 'Functional Units', with the exception of fuel. There was no MGA on the fuel weights. Based on the standard, this is conservative for most of the large weight items. At Layout, structure and propulsion are both 15% MGAs in the standard.

F. **Cost Model**

The cost model came completely from reference [27] by Matthias Grosse. It's based on the production of costs of the weight of an individual piece part based on historical precedence. Gross defined Functional units as listed in **Table 5**. Each of those functional units were assigned cost numbers.

Table 5 Components Sorting Scheme and cost index for Boosters Data Base[27]

Functional Unit	Related Component	Cost Index (Cost Unit/kg)
"Equipment"	Power supply, harness, instrumentation, telemetry, commando unit, rocket motors for stage separation, pyrotechnics for separation and self-destruction	17
"Tank"	Equipped liquid propellant or oxidizer tank: Tank structure, isolation, propellant pipes, antivortex and -sloshing devices and tank pressurization system (not part of engine or LOX feed unit)	6
"Motor Case"	Rocket motor case incl. insulation, liner and igniter for solid fuel/propellant	1
"Nozzle"	Solid rocket like ablative nozzle with hydraulic actuated thrust vector control unit	4
"Engine" / "LOX Feed Unit"	Liquid rocket engine (incl. Actuation system and control units) or technological comparable "LOX Feed Unit" of the hybrid rocket (turbopump, injector, valves, gas generator and its fuel tank)	20
Solid Propellant		0.1
Hybrid Propellant		0.05

The Grosse analysis relied on the concept that hybrids are a combination of liquids and solids components, so scale factors could be used to scale the boosters. Mass indices for various components were taken from solid motors, liquid rockets and previous hybrid studies. The previous hybrid studies included AMROC's H-1800 motor, HPDP's 250K and Lockheed Martin's Falcon Upper Stage Demonstrator. Grosse wrote: "The solid and liquid rocket reference booster models rely on data from the Ariane 5 solid rocket booster EAP, from the Ariane 5 liquid booster study for the proposed EAL (Etage d' Accélération à ergols Liquides) using kerosene as fuel, and from the Ariane 4 liquid booster L36 and its second stage L33. Schmucker [5] has determined that the most cost effective design is the use of a liquid engine with a lower chamber pressure for the first stage or a booster. Therefore, for the liquid propellant reference booster, a hypothetical liquid LOX/kerosene rocket engine similar in Isp and T/W-ratio to the F-1 engine is foreseen. …."[27] The "Schmucker [5]" reference is not in English and wasn't reviewed as part of this effort and the data was used directly from the Grosse paper.

Grosse used the scaling equation to generate a mass estimate of the solid, liquid and hybrid boosters. He compared them back to other references "for model verification, the ratio of manufacturing cost to fueled mass of the liquid and solid rocket booster was evaluated. The calculated cost ratio of 2.95 is comparable to a reference value of 2.76 [Wells] and to results found in [Roberts]." Those references have not been reviewed yet, so an independent check has not been performed.

Table 6 Mass Data of Single Boosters and their Units – Grosse[27]

Fvac/mo=2.6	Liquid	Solid	Hybrid Baseline

Launch Mass(t)	206	292	335
Structural Index	0.0980	0.1596	0.1534
Functional Unit	Mass(t)		
“Structure”	5.1(28%)	5.3(13%)	8.5(19%)
“Equipment”	1.5(8%)	2.0(5%)	2.3(5%)
“Tank”	6.3(34%)	N/A	5.1(12%)
“Motor Case”	N/A	22.9(57%)	14.8(33%)
“Nozzle”	N/A	9.9(25%)	11.6(26%)
“Engine/Lox Feed Unit”	5.5(30%)	N/A	2.1(5%)
Inert Mass	18.4	40.1	44.4

Using the mass estimates from Table 6 and the functional cost from **Table 5**, Grosse calculated the cost of the boosters. The calculations show that in terms of cost (lowest to highest), the order is solids, liquids and then hybrid boosters. This is different from the traditional hybrid rocket paradigm, where hybrids are cheaper. This was the rationale for conducting the new analysis.

Table 7 Cost Distribution between Functional Units – Grosse[27]

Functional Unit	Liquid	Solid	Hybrid
“Structure”	11%	15%	16%
“Equipment”	13%	23%	19%
“Tank”	19%	N/A	14%
“Motor Case”	N/A	16%	7%
“Nozzle”	N/A	28%	22%
“Engine/Lox Feed Unit”	57%	N/A	20%
Inert Mass	N/A	18%	2%
Total Booster cost (derived)	193.7	142.7	210.5
Total Booster, relative	135%	=100%	149%

Given that the analysis rated hybrids so costly via this analysis, it seemed like a fair approach to generate a cost based on a bottoms up approach. Having a system where there is a liquid and solid rocket booster cost is very convenient for comparison analysis.

Grosse's paper points out that that there are many costs in the use of a booster system. "As stated by Koelle[40], typically 75% of launch cost comes from the fabrication, assembly and verification of vehicle elements. Ground and launch pad operations to assemble, checkout, transport, tank and fill the vehicle, together with the launch and flight operations to plan, control, track and assess its flight account for 15%. The remainder will be caused by the management, marketing, customer relation, contracts office, technical support and launch site costs." In order for hybrids to be competitive, the fabrication, assembly and verification portions of a hybrid booster need to be lower than a solid or liquid booster. Operations costs, say the explosive potential of a solid or loading of two fuels for a liquid, are small pieces of the total cost, so savings there wouldn't drive the costs. However, if savings are realized in the fabrication, assembly and verification parts, there should be savings in the launch and flight operations.

IV. Results

Several different combinations were looked at: An AMROC approximation, A baseline Lox Polybutadiene Booster, a lox aluminized Polybutadiene Booster.

A. An AMROC approximation

For comparison purposes, two cases were run with the burnout of the grain occurring at the last segment, both with one row only allowed. The center port was allowed to burn, where the AMROC center port was blocked. One was set at as close to the same conditions as the AMROC 250k condition, with the same number of ports, similar pc, flux, etc. This should be close to what Grosse was using for the AMROC and HPDP 250K comparison. In **Table 8 Ariane Solid vs Minimum Cost Booster LOX Polybutadiene with Nsegchk=9**, the values for booster cost bracket what Grosse calculated for a scaled up booster. The lengths and diameters are quite large compared to the baseline Ariane solid booster and the core. The costs are cheaper than Gross solution. The 9 port solution is close to a match on the diameter of the solid booster, but significantly longer.

The diameter of the motor makes it a nonstarter to replace the solid, however as an example of the model output, the grain configuration is shown in **Figure 24 AMROC 250K scaled to Booster size**. The light gray thicker part of the spoke is the preborn condition, the darker center represents the burnout condition. In this case, with Nsegchk=9, the last of the fuel grain segments have burned out but the forward sections forward of that remain. In this case, the thickness in the forward end of the the grain is ~.9 inches. This makes for a long thin grain segments that could burn thru or flex to failure.

Figure 23 AMROC 250K scaled to Booster size shows the pressure and thrust output of the model. With nsegchk=9, the last segment burns out before the end of burn. That burnout is shown in the pressure and thrust drop near the end of burn.

Table 8 Ariane Solid vs Minimum Cost Booster LOX Polybutadiene with Nsegchk=9

	Solid (P240 Ariane)[41]	Grosse Hybrid Solution [27]	Hybrid (1 row) Nsegchk=9	Hybrid (1 row) Nsegchk=9 forced to 15 ports AMROC
Ports/Rows			9 P / 1 R	15 P / 1 R
Booster diameter(ft)	10.00		10.6	17.5
Booster length(ft)	103.6		224.7	173.2

Booster gross mass lb	618000	648,256	883248	1,053,740
Booster dry wt (no lox) lb	n/a		368206	461,924
Thrust Lbf (average)	1,140,000		1,849,407	2,287,349
Ave Vac ISP(sec)	275.4	278	283	295
Cost (cost units)	142,700[27]	210,500	117,426	159,205
Residual fuel %			8.5	18.5

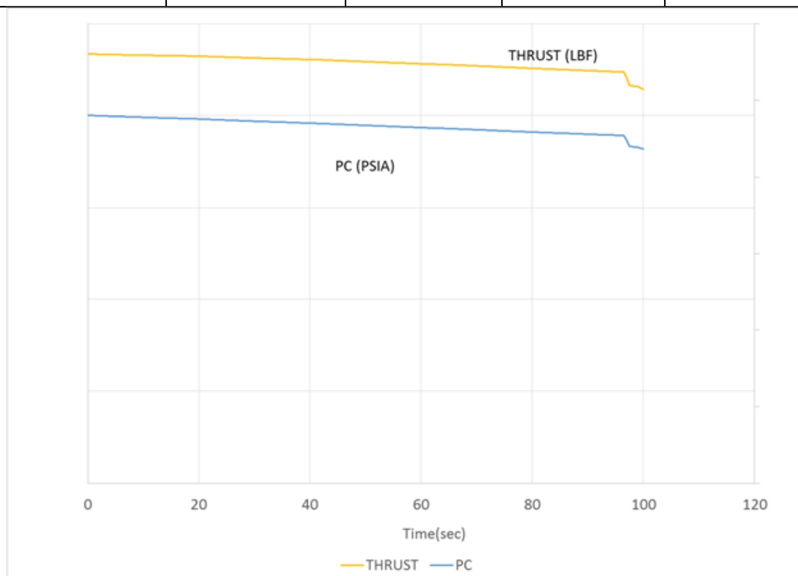

Figure 23 AMROC 250K scaled to Booster size performance

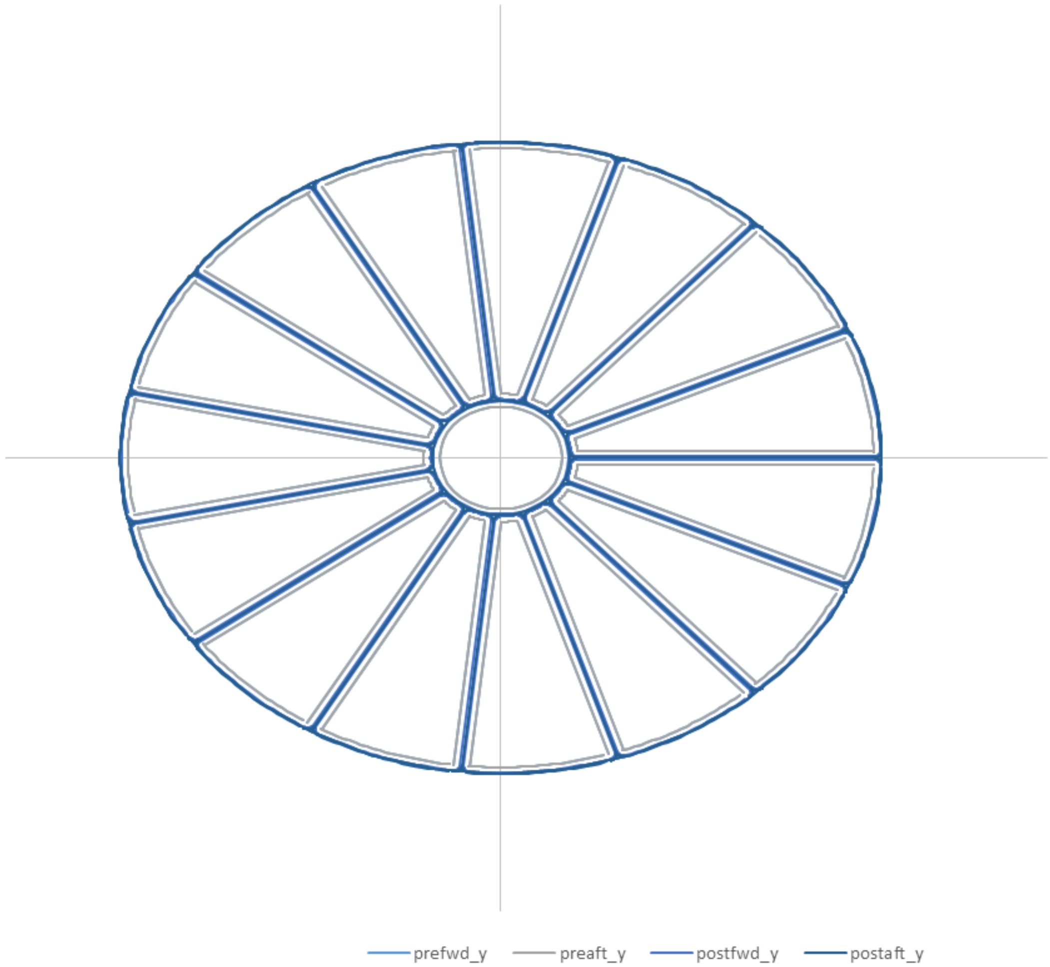

Figure 24 AMROC 250K scaled to Booster size geometry

As shown, these are large ports, larger than what was used during the AMROC and HPDP 250K testing. There may be a correlation between larger ports and lower regression given the same oxidizer flux[42]. Yee indicated there was a 30% drop off in the performance of the HPDP 250K motor using a burn rate from from a ¾ scale port test compared to the smaller multi port motor that was used to derive the burnrate used in the design.[43]. A function is included to account for that correlation to diameter in the model, however it is an extrapolation from the largest motors fired and unverified.

B. Baseline Lox Polybutadiene Hybrid Booster

Table 9 Lox Polybutadiene boosters minimizing on cost

	Solid (P240) Ariane	Grosse Hybrid Solution	Hybrid nsegchk=9	Hybrid nsegchk=5	Hybrid nsegchk=3
Ports/Rows			5 P / 7 R	5 P / 7 R	5 P / 7 R

Booster diameter(ft)	10.00		15	14.2	13.9
Booster length(ft)	103.6		98.6	97.2	98.3
Booster gross mass lb	618,000	648,256	696,731	633,483	625,860
Booster dry wt (no lox) lb	n/a		254,020	227,886	223,873
Thrust Lbf (average)	1,140,000		1,331,564	1,253,613	1,228,307
Ave Vac ISP(sec)	275.4	278	283.5	286.7	283.0
Cost (cost units)	142,700[27]	210,500	98,653	92,820	91,127
Residual fuel %			20.0	14.4	12.6

Table 9 Lox Polybutadiene boosters minimizing on cost, with 7 rows, the boosters optimize on cost to short fat boosters. Earlier in the analysis process with only 3 rows available, the genetic algorithm was optimizing out with 2 or three rows of ports. For the boosters that optimized with 3 rows of ports, it was unclear if 3 was the correct answer or just the upper limit. Later in the evaluation process and paper writing process, it was decided to modify the function that was being evaluated from total cost to total cost and booster length. A longer booster would could be an issue in an assembly building, well, exiting the vehicle assembly building. The down selected function to minimize included total cost and length as equal parameters, evaluated against the Ariane Solid booster parameters. This lead to boosters that were always optimizing out at 3 rows. More rows were added to the model, so 7 burning Rows were available. The rest of the analysis includes the option for 7 rows of ports.

As can be seen in **Table 10-Lox Polybutadiene Minimizing Cost and Booster Length**, the costs and lengths are in line with or lower than the costs of the solid and substantially lower than the Grosse hybrid cost. However, the booster diameter is much higher, which would be a drag issue for the vehicle.

Table 10-Lox Polybutadiene Minimizing Cost and Booster Length

	Solid (P240 Ariane)[44]	Grosse Hybrid Solution [27]	Hybrid Nsegchk=9	Hybrid Nsegchk=5	Hybrid Nsegchk=3
Ports/Rows	n/a		6 P / 7 R	6 P / 7 R	8 P / 7 R
Booster diameter(ft)	10.00		15.8	14.3	14.6
Booster length(ft)	103.6		97.4	97.3	93.5

Booster gross mass lb	618000	648,256	486,356	658,516	448,667
Booster dry wt (no lox) lb	n/a		260,464	238,607	243,548
Thrust Lbf (average)	1,140,000		1,351,437	1,275,612	1,294,508
Ave Vac ISP(sec)	275.4	278	283.3	281.6	280.9
Cost (cost units)	142,700[27]	210,500	99,553	94,049	93,541
Residual fuel %			19.1	13.8	12.2

Figure 25 Polybutadiene LOX nsegchk=3 min cost and booster length performance shows the predicted performance of the booster. The tailoff shows the burnout of the last 7 axial stations.

Figure 26 LOX/Polybutadiene nsegchk=3 min cost and booster length grain configuration shows the grain configuration at ignition and at shutdown. The minimum web thickness is on the order of 0.3 inches, with short segments. If the tensile properties of the fuel can reach those shown in **Figure 18 Tensile Strength and Strain of LMF-900 Fuel vs. Previous Fuels**, there may be sufficient strength to hold the structure together, however a structural analysis was not done as part of this study. Assuming fairly uniform burning, if one of the web sections fail, it shouldn't be as bad as if one of the webs fails in **Figure 24 AMROC 250K scaled to Booster size geometry**. Also, the smaller ports should yield fuel regression rates closer to the base burnrate being used.

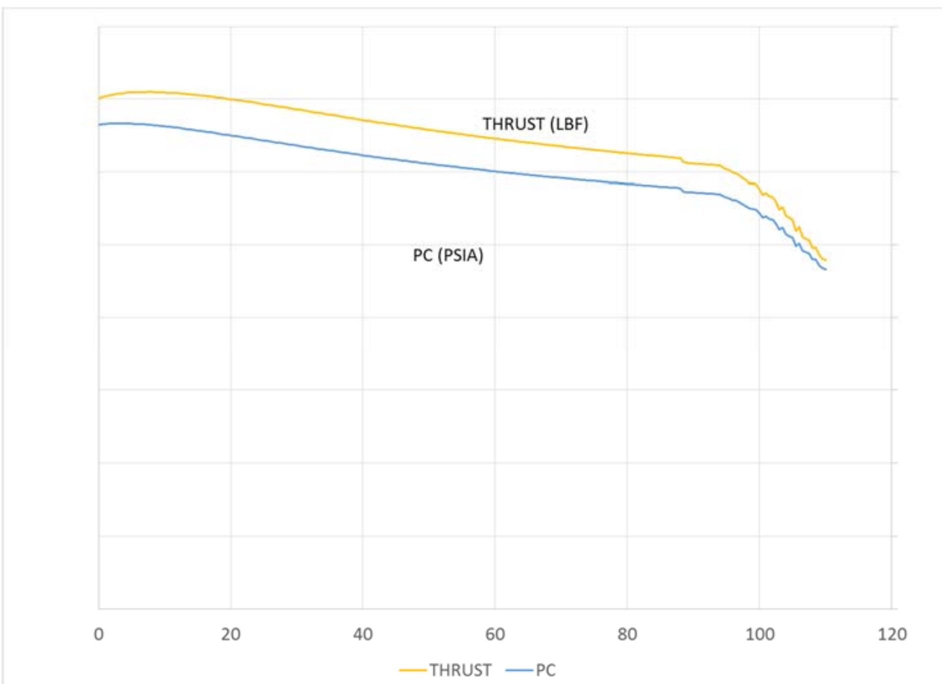

Figure 25 Polybutadiene LOX nsegchk=3 min cost and booster length performance

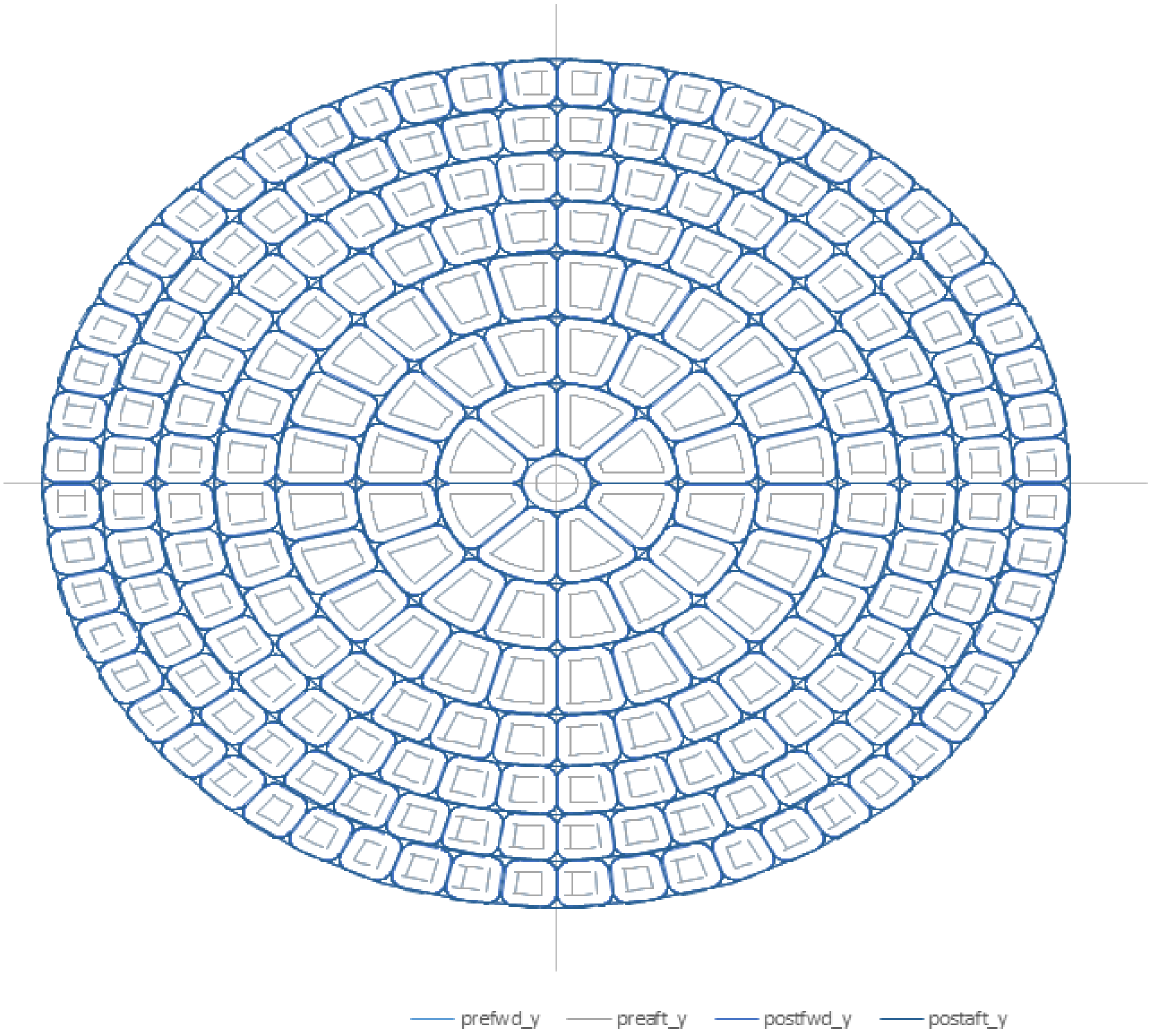

Figure 26 LOX/Polybutadiene nsegchk=3 min cost and booster length grain configuration

Figure 26 LOX/Polybutadiene nsegchk=3 min cost and booster length grain configuration shows the burn out configuration for the motor. One of the concerns about hybrids is the residual fuel weights[45]. Lockheed Martin's solution for the fuel residual problem is hybrids[29] was to direct the oxidizer to 'burn the motor preferentially from the center row out.' For this exercise, a different tack was taken, where allowed the natural tendency for more fuel regression at the aft end of the motor to work its way up the motor case. At the end of burn, the forward end of the motor is approximately 0..35 inches thick, tapering to burnt out at the midway point in the motor case, with the rest of the fuel (except for the slivers) consumed during the burn. This also reduced the residual fuel weight, however there is still a large amount remaining, ~12%, which is much more than the Lockheed Martin planned 3%. The vehicle size is much larger than it could be with a lower residual weight, however in it's current configuration, it is cheaper than the solid or liquid booster. Currently the code stops burning at an axial station, the slivers in that section stop burning and the slivers left are along for the ride as inert weight. This physically is not the case as can be shown in **Figure 17 RR102+ Post Test Photos and Chamber Pressure**; in that case they should be stuck out from the forward end since it regressed faster than the aft end in that motor configuration. Modifying the code to continue to burn exposed slivers could help with some of that residual weight and the Lockheed Martin inside out approach could be a future modeling effort. Either approach relies on high tensile strength fuel to deal with the thin webs towards burn out.

Due to vehicle constraints, long narrow boosters might not work and short fat boosters may not work either. An attempt was made in the optimization function to force the booster diameter to 10 ft. See **Table 11 Optimizing with LOX/Polybutadiene 10 ft dia case**. As a first run, the code optimized out to a 1 row configuration, which was really long booster. Playing with the 10 foot diameter forcing function and forcing to a one row connection led to conditions where the booster was hitting upper limits on some of the variables and, with the current set of burn rate and port flow assumptions, led to boosters that were closer to 10 ft diameter, still smaller than the previous analyses, but larger than the solid booster. The booster costs are still considerably less than the stated solid costs or the hybrid solution from reference [27].

Table 11 Optimizing with LOX/Polybutadiene 10 ft dia case

	Solid (P240 Ariane)[46]	Grosse Hybrid Solution [27]	Hybrid nsegchk=3 Open, chose 1 row	Hybrid nsegchk=3 Open, chose 2 row	Hybrid nsegchk=3 Force to 4 row	Hybrid nsegchk=3 Force to 3 row	Hybrid nsegchk=3 Force to 7 row
Ports/Row			P 7 / 1 R	5 P / 2 R	5 P / 4 R	5 P / 3 R	5 P / 7 R
Booster diameter(ft)	10.00		9.92	9.99	12.39	10.94	13.39
Booster length(ft)	103.6		283.8	217.4	143.3	176.1	100.3
Booster gross mass lb	618,000	648,256	819,573	789,075	728,195	791,449	661,410
Booster dry wt (no lox) lb	n/a		342,274	328,385	292,758	331,825	239,679
Thrust Lbf (average)	1,140,000		1,713,878	1,655,982	1,548,853	1,692,621	1,358,030
Ave Vac ISP(sec)	275.4	278	271.9	269.1	274.3	271.0	288.1
Cost (cost units)	142,700[27]	210,500	119,985	111,409	102,071	110,938	98,388
Residual fuel %			6.1	7.9	9.6	9.1	16.1

C. Lox/Polybutadiene/AL Hybrid Booster

For the aluminized propellant, only one aluminum loading was looked at, 25%. There was not a detailed examination into the loading level. High levels of solids could lead to lower tensile properties of the fuel, which may be counterproductive to the multiport multi row thin web design.

Table 12 Lox/Polybutadiene/Al Runs

	Solid (P240 Ariane)[47]	Grosse Hybrid Solution[27]	Hybrid Nsegchk=9	Hybrid nsegchk=5	Hybrid nsegchk=3
Ports / Rows			5 P / 7 R	4 P / 7 R	8 P / 7 R
Booster diameter(ft)	10.00		17.9	16.2	17.5
Booster length(ft)	103.6		93.1	89.1	86.4
Booster Gross mass lb	618,000	648,256	946,187	808,213	838,604
Booster dry wt (no lox) lb	n/a		420,424	355,133	385,524
Thrust Lbf (average)	1,140,000		1,597,305	1,313,530	1,247,528
Ave Vac ISP(sec)	275.4	278	290.7	286.2	286.9
Cost (cost units)	142,700[27]	210,500	118,321	101,407	100,791
Residual fuel %			25.9	24.5	22.0

The aluminum loading has a double effect on performance that makes for a bigger booster. The first effect is a shift down in peak oxidizer to fuel ratio compared to the non-aluminum loaded fuel, so a higher fuel flow is necessary. Some of that can be made back from the higher regression rates with aluminized fuels and the higher density. The second effect is that the residual fuel volume weighs more. The net effect is that, with the current setup of the model, the aluminum loaded fuel booster optimizes a larger diameter but shorter booster than the nonaluminum loaded booster. The computational remaining sliver issue, where the fuel slivers in burnt out fuel section are no longer considered burning, hurts the aluminized grains more. Future work should develop a method to burn those slivers and lower the residual weights.

V. Summary and Conclusions

1) This analysis has shown that, given the assumptions in the analysis, the cost of a hybrid rocket booster for this application is equal to or lower than the cost of a solid or liquid rocket booster. This is different than the results of the Grosse analysis. An explanation

for the difference in conclusions is Grosse used the extrapolation of point design to a much larger size.

2) A LOX/Polybutadiene hybrid booster is still larger than a solid or liquid booster for the same application. Future designs should include requirements based physical limits of the vehicle assembly building, launch vehicle configuration, etc.

3) Residual weigh of the fuel grain is still a design issue for a hybrid booster. Accounting for the residual can lead to large boosters. This analysis leaves the classical fuel slivers in the 'burnt out' sections unburnt and is considered as inert mass, detrimental to the performance of the booster. Analytically modeling it to burn and accounting for this will reduce the size and weight of the booster predictions.

4) The multiport multirow grain design has some limited testing that demonstrated the concept. In this analysis trying to match the 10 ft diameter, some of the designs with lower number of rows of ports were able to meet 10 ft diameter, however these grains had larger ports than previously demonstrated.

5) Extrapolation of burn rates from small ports to large ports has been shown to overestimate the regression rate of the large port. Designs that greatly exceed those sizes have regression rates that have been extrapolated from existing data. Extrapolation of the burn rate to that point may or may not be accurate and needs to be verified in single port firings before building larger hardware.

VI. FUTURE WORK

Modification of the optimization parameters – The final optimization parameter used in this analysis was based on minimizing total cost and booster length. An alternate analysis should be done on total cost and booster diameter, since the drag of the booster is probably a function of diameter instead of length. Another alternative is basing the length on the height of the assembly building doors.

Pressure Fed options – One of the cost drivers in the hybrid system modeled is the pressurization system, that is the turbopumps. Per the baseline cost indices, the turbopumps are 20 cost units/kg vs tanks at 6 cost units/kg. The tank size should be roughly the same, with or without the turbopump, the difference being in the tank thickness. If the cost of the increase in tank mass is less than the cost of the turbopump, this concept could be a winner. Tridyne was originally developed by Rocketdyne for tank pressurization[48]. It consists of "a single storage tank containing a nondetonatable mixture of an inert gas, an oxidizer and a fuel. A catalytic bed is functionally connected therewith whereby the oxidizer and fuel are ignited by the catalytic bed, producing hot gases." Work by AMROC on the SET-1 Flight vehicle used a Triadyne pressure fed system with a separate oxidizer tank in the flight system.[49] Recent work in pressure fed systems has demonstrated an improvement of a Tridyne pressurization system[50], where the catalyst bed is suspended in the Triadyne tank. This allows the heat from the catalyst bed to also heats the pressurant remaining in the tank, increasing expulsion efficiency. Their analysis indicates a 50% decrease in pressurant mass vs a cold gas system.

Detailed Trajectory analysis – The analysis as completed does not do any trajectory analysis to gage the performance of the hybrid system. A simple average ISP and rocket equation were used to do a basic analysis of the hybrid system. However, this approach does match well with the baseline comparison analysis done by Grosse[27]. Potential future work includes coupling the hybrid code with a launch performance code, similar to previous work done on other hybrid sizing analysis[51], except optimizing on cost basis.

Other oxidizer/fuel combinations – Much work has been done on the development of liquefying hybrid rocket propulsion (paraffin hybrids) and alternate oxidizers. The use of a paraffin hybrid, with its high regression rate, would greatly reduce the residual propellant in the motor case at burnout. That lower burnout weight should result in a smaller sized booster. Also, the use of Nitrous Oxide, Nitrox or Hydrogen Peroxide could simplify the propulsion system since they are non-cryogenic. The peak ISP for these oxidizers occurs at a higher O/F then for a lox based system, which means less fuel is required and therefore less residual propellant left after motor shutdown.

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