Genetic Algorithm Optimization of a Cost Competitive Hybrid Rocket Booster

George Story

NASA MSFC, Huntsville Al 35812, Al

Abstract

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 later on solid rockets. Slowly the technology readiness level of hybrids has been increasing due to various large scale testing and flight tests of hybrid rockets. A 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.

Nomenclature

Introduction

A. Motivation

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, and hybrids may be getting renew interest. Hybrids are soon to fly on Virgin Galactic's SpaceShipTwo and were on Sierra Nevada's Dream Chaser vehicle initially. 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.

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 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 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 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 Hydroxyl Terminated Polybutadiene - LOX rocket is equivalent to a RP-1-LOX engine, and significantly higher than a solid rocket motor which is ~40 seconds higher than a HTPB/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 does it make financial sense to develop them for launch vehicles. AMROC, during the 1990, developed hybrids with commercial venture funds.

A recent top level study, "Design Challenges for a Cost competitive Hybrid Rocket Booster"[3], 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.

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[Error! Bookmark not defined.,3,1]. For this paper, we'll review concepts that are relevant to the design envelope being discussed here.

Approach and Models

A. Reference Booster and Mission

The reference mission and cost reference data is based on a comparison study by Grosse[3]. 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.

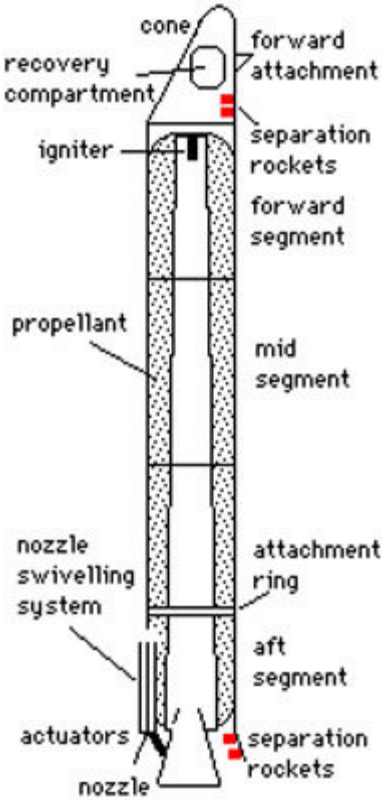

Figure 1Ariane Launcher Booster[2]

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)."[3]
-Solid Reference Booster is based on Ariane 5 EAP.
-Liquid Reference Counterpart is based on Astriums' EAL study + F-1 type engine to replace RD-180.

B. Configuration of the Hybrid Booster

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 HTPB combination and a LOX HTPB 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.

c. Number of rows – This affects the volumetric efficiency of a booster. Based on previous analysis in Ref 5, the change from 3 to 4 rows doesn't lessen the void space greatly, so the model is limited from 1 to 3 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 - 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[4]. 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 5. 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[5], 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 gass 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 I'd heard 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.

2) 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 3 rows of ports, depending on the input.

b. The forward and aft domes are fuel lined ¾ ellipses with a ½ inch layer of silica phenolic as 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[14] empirical nozzle sizing calculations in section 7.6.4.

e. The TVC weight is just an approximation, assuming ½ the nozzle weight per Humble[14] 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 silica phenolic as 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 [14] 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 [6] and [5]. 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 [5], 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.

3) 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 out from the inside out. When the fuel web was a mininmum 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. 'Nsegchk' is a 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.

C. 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."[7] The particular Genetic Algorithm code being used was downloaded from the web [reference 7] in the late 1990s after reading another paper on genetic algorithms used in the development of hybrid rocket motor designs.[8] Genetic algorithms have been used to size multiple rocket configurations.[9],[10],[11],[12],[13]

2) Summary of Code - The basic code flow was copied from Reference [8]. 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 2 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 motor. 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[14]) and others just rough estimates (TVC weight is ½ weight of nozzle, Humble 6.3.8). The code converges on the hybrid motor 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, seeTable 1. As shown, difference is numbers is small compared to the rough estimate of the analysis.

Table 1 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

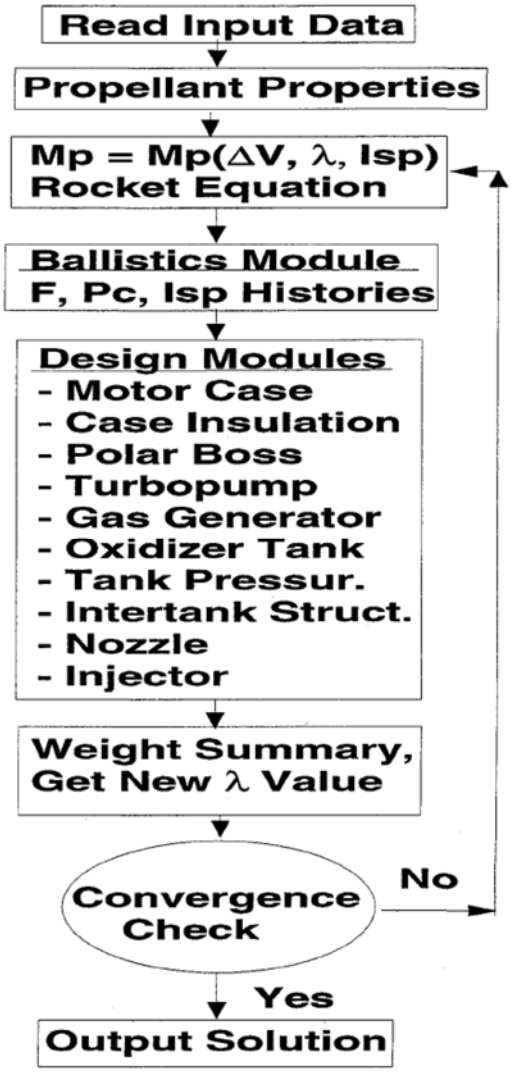

Figure 2 Flowchart for HYROCS code[8]

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[15] 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.
5) Cost Model – The cost model came completely from reference [3] 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 2.

Table 2 Components Sorting Scheme for Boosters Data Base

Functional Unit	Related Component
"Structure"	Nose cone, forward skirt, rear skirt, heat shield, interstage section, intertank section, forward and rear attachment system, pipe and harness ducts
"Equipment"	Power supply, harness, instrumentation, telemetry, commando unit, rocket motors for stage separation, pyrotechnics for separation and self-destruction
"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)
"Motor Case"	Rocket motor case incl. insulation, liner and igniter for solid fuel/propellant
"Nozzle"	Solid rocket like ablative nozzle with hydraulic actuated thrust vector control unit
"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)

Each of those functional units were assigned cost numbers.

Table 3 Cost Indices of Functional Units - Grosse

Functional Unit	Cost Index (Cost Unit/kg)
"Structure"	4
"Equipment"	17
"Tank"	6
“Motor Case”	1
“Nozzle”	4
“Engine” / “LOX Feed Unit”	20
Solid Propellant	0.1
Hybrid Solid Fuel	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 motors. 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. ….”[3] 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 4 Mass Data of Single Boosters and their Units – Grosse

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 4 and the functional cost from Table 3, 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 5 Cost Distribution between Functional Units – Grosse

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 motor 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[16], 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.

Results

Several different combinations were looked at: A baseline Lox HTPB Booster, a lox aluminized HTPB Booster.

A. Baseline Lox HTPB Hybrid Booster

With the genetic algorithm searching based on cost to meet the delta velocity requirement, it selected a series of inputs with the lowest total cost. As an example of how the genetic Algorithm works thru the process, there are some values that are quite high, see Figure 3. Those characteristics are quickly eliminated as the generations are progressed thru. At certain points in the process, the genetic algorithm rescrambles the characteristics to ensure that the minimum solution isn't missed.

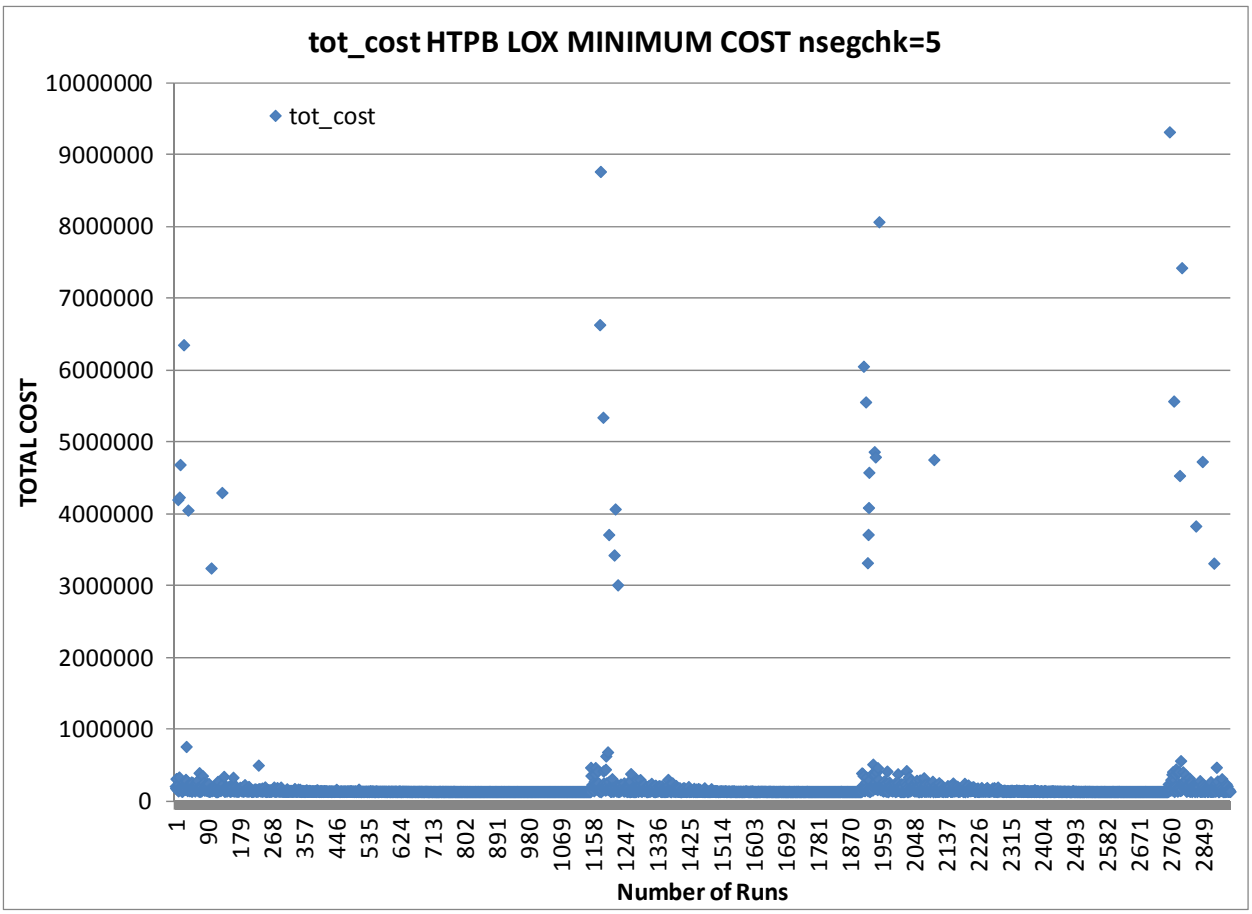

Figure 3 Total Cost HTPB LOX Minimum Cost

Zooming in closer in Figure 4, there are lots of solutions that approach 140,000 cost units. Some of the values however, are in a thrust to weight level that is too low for this comparison exercise.

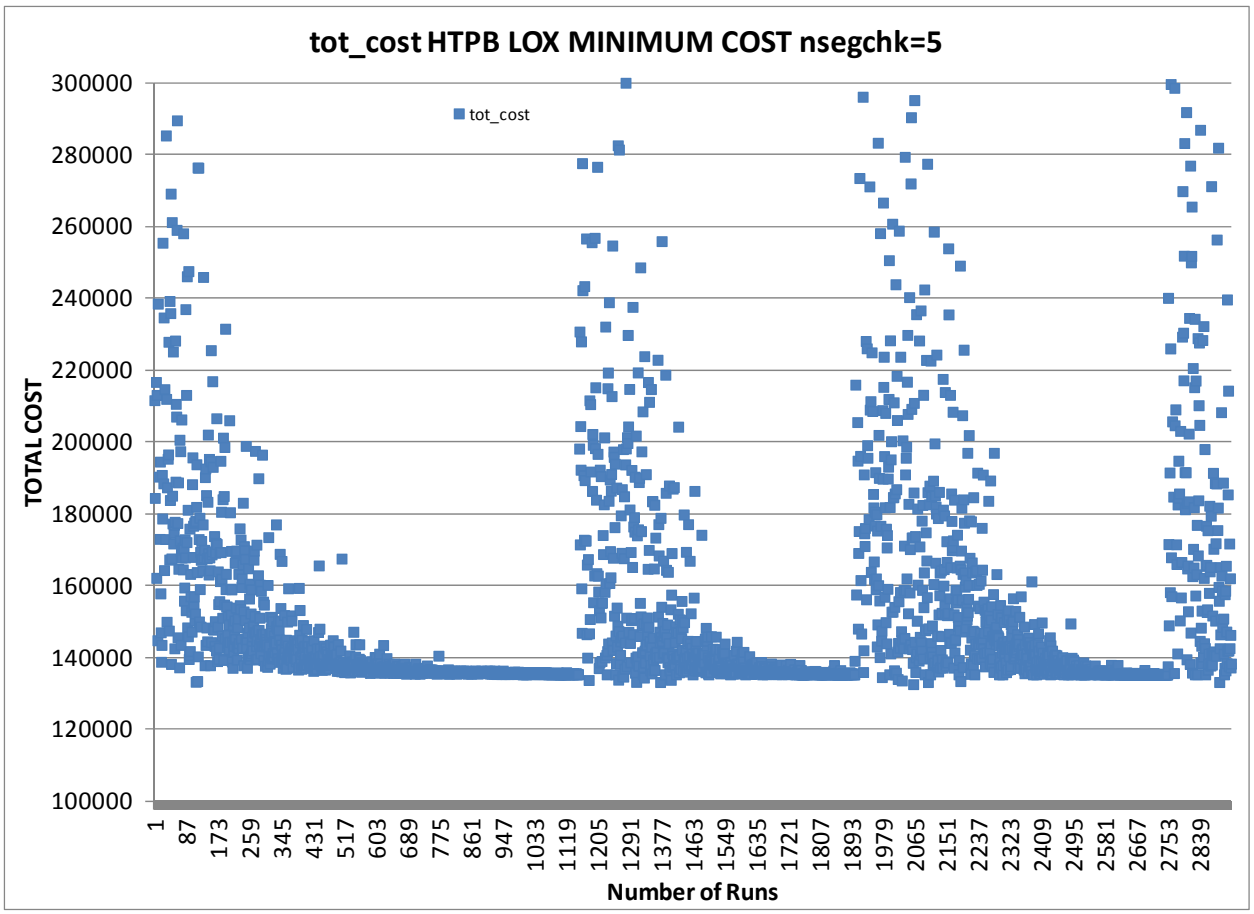

Figure 4 Tot_Cost HTPB LOX Minimum Cost Zoomed in

Looking at the data compared to vacuum thrust/weight shows distinct families of total costs. All of the combinations shown will reach the desired deltavelocity, however the different burn times affect the thrust to weight. A shorter burn time, with the same basic impulse will drive to a higher thrust to weight, which drives a higher oxidizer flow rate and larger pumps and higher costs.

For comparison purposes, a case was run with the burnout of the grain occurring at the last segment. This should be close to what Grosse was using for the AMROC and HPDP 250K comparison. The values are all from one run, however the different rows are shown to point out the effect of one row (tier 1), two rows (tier 2) or three rows (tier 3) ports. While an individual run for one (tier 1) or three (tier 3) rows might give a slightly different answer, the trends are that there is a large range of total cost values at the same thrust to weight. Some of the values are above the Grosse estimate, some are below the estimate. My theory is that the Grosse answer is an extrapolation of a hybrid grain to another size and while representative of the cost of a design, it is not indicative of a minimum cost for a design.

In another point of that is rather obvious is that most of the hybrid solutions at at thrust to weight of 2.6 are below the Grosse Liquid and Hybrid costs but above the cost of the solids.

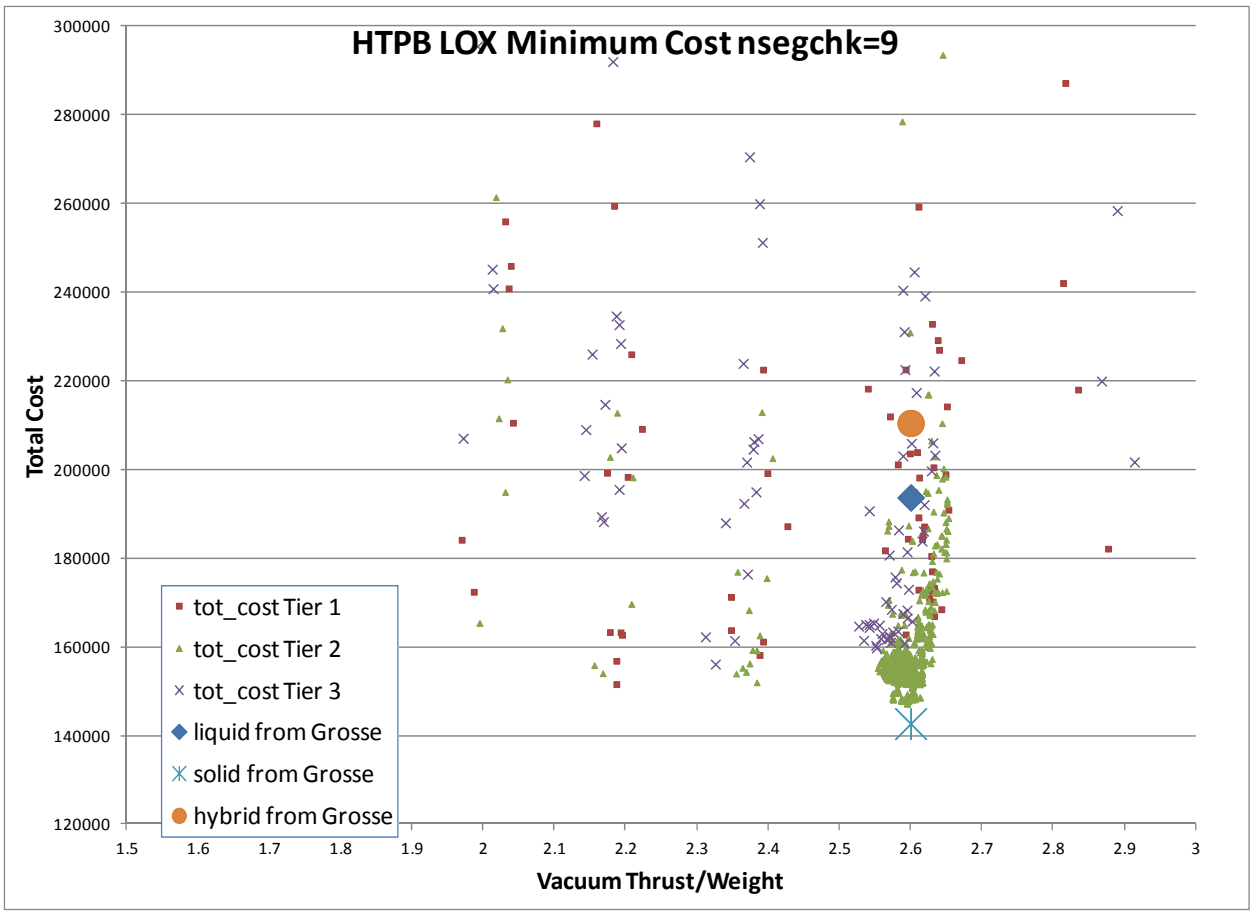

Figure 5 Total Cost HTPB LOX Minimum Cost vs thrust/weight

One of the concerns about hybrids is the residual fuel weights. Lockheed Martin's solution for the fuel residual problem is hybrids[5] 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 0.7 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, ~13%, 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. 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.

As can be seen in Based on the sensitivity analysis in Table 1, the costs are essentially equivalent.

Table 6, the hybrid solution is considerably larger than the Solid Booster in all parameters except for burn time and cost. Based on the sensitivity analysis in Table 1, the costs are essentially equivalent.

Table 6 Ariane Solid vs Minimum Cost Booster LOX HTPB with Nsegchk=9

	Solid (P240 Ariane)[17]	Hybrid Solution [3]	Hybrid (1 tier) Nsegchk=9	Hybrid (2 tier) Nsegchk=9	Hybrid (3 tier) Nsegchk=9
Booster diameter(ft)	10.00		11.4	13.4	14.2
Booster length(ft)	103.6		312	191	178
Booster Gross mass lb	618000		1,226,369	1,115,692	1,247,829
Booster dry wt (no lox) lb	n/a		508,020	484,424	529,660
Thrust Lbf (average)	1,140,000		2,602,847	2,577,606	2,723,570
Ave Vac ISP(sec)	275.4		239.9 SL	245.2 SL	250.0 SL
Burntime(sec)	130		100	100	100
Cost (cost units)	142,700	210,500	162807	147,239	160,793

B. Aluminized Lox HTPB AL Hybrid Booster

	Solid (P240 Ariane)[18]	Hybrid (1 tier) Nsegchk=5
Booster diameter(ft)	10.00	11.3
Booster length(ft)	103.6	224
Booster Gross mass lb	618000	1052474
Thrust Lbf (average)	1,140,000	24941111

Ave Vac ISP(sec)	275.4	283.9
Burntime(sec)	130	110
Cost (cost units)	142,700	131202

C.

Future work

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. 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[19]. It consists of "a single storage tank containing a nondetonatable mixture of an inert gase, 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.[20] Recent work in pressure fed systems has demonstrated an improvement of a Tridyne pressurization system[21], 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 a sophisticated 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[3]. Potential future work includes coupling the hybrid code with a launch performance code, similar to previous work done on other hybrid sizing analysis[22], except optimizing on cost basis.

Varied inputs

1. **Target O/F, etc**

Other oxidizer/fuel combinations – Much work has been done on the development of liquefying hybrid rocket propulsion (AKA paraffin hybrids) [reference Arif/Greg's papers] 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 [Reference Arif] or Hydrogen Peroxide [Reference Heister papers] 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.

Conclusions

1) **This analysis has shown that, given the assumptions in the analysis, the cost of a hybrid rocket motor for this application is equal to or lower than the cost of a solid or liquid rocket motor. This is different than the results of the Grosse analysis. A possible explanation for the difference in conclusions is the extrapolation of point designs to a much larger size.**

2) **A lox/htpb hybrid motor is still much larger than a solid or liquid motor for the same application.**

---

[1] Chiaverini, M.J., Kuo, K.K.,"Fundamentals of Hybrid Rocket Combustion and Propulsion", AIAA Progress in Astronautics and Aeronautics, Vol 218, 2007.

[2] http://www.esa.int/Our_Activities/Launchers/Launch_vehicles/Boosters_EAP
Launch Vehicles, Launcher, accessed 12 29 2014

[3] Grosse, Matthias, 'Design Challenges for a Cost Competitive Hybrid Rocket Booster', Presented at EUCASS, 2nd European Conference for Aerospace Sciences, Brussels, July 1-6, 2007

[4] Boardman, T., Abel, T., Claflin, S, Shaeffer, C., "DESIGN AND TEST PLANNING FOR A 250-KLBF-THRUST HYBRID ROCKET MOTOR UNDER THE HYBRID PROPULSION DEMONSTRATION PROGRAM", AIAA 1997-2804

[5] Kearney, D., Joiner, K., Gnau, M. and Casemore, M., "Improvements to the Marketability of Hybrid Propulsion Technologies", AIAA SPACE 2007 Conference & Exposition, AIAA 2007-6144, 18 - 20 September 2007, Long Beach, California

[6] Dupuis, T., Knowles, T, "Oxygen Rich Hybrid Gas Generator", 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, AIAA 2006-4673, 9 - 12 July 2006, Sacramento, California

[7] http://www.cuaerospace.com/carroll/ga.html, FORTRAN Genetic Algorithm (GA) Driver, David L. Carroll
CU Aerospace, 2100 South Oak Street, Suite 206, Champaign, Illinois 61820

[8] Schoonover, P. L., Crossley, W. A. , Heister , S. D., "APPLICATION OF GENETIC ALGORITHMS TO THE OPTIMIZATION OF HYBRID ROCKETS", AIAA 1998-3349, AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 34th, Cleveland, OH, July 13-15, 1998

[9] Riddle, D.B., Harfield, R.J., Burkhalter, J.E. and Jenkins, R.M.,"Genetic Algorithm Optimization of Liquid Propellant Missile Systems", 45th AIAA Aerospace Sciences Meeting and Exhibit, AIAA 2007-362, 8 - 11 January 2007, Reno, Nevada

[10] Metts, J.G., Hartfield, R.J., Burkhalter, J.E. and Jenkins, R.M., "Reverse Engineering of Solid Rocket Missiles with a Genetic Algorithm", 45th AIAA Aerospace Sciences Meeting and Exhibit, AIAA 2007-363, 8 - 11 January 2007, Reno, Nevada

[11] Bayley, D.J. and Hartfield, R.J," Design Optimization of Space Launch Vehicles for Minimum Cost Using a Genetic Algorithm", 43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, AIAA 2007-5852, 8-11 July 2007, Cincinnati, Ohio

[12] Bayley, D.J., Hartfield, R.J., Burkhalter, J.E., and Jenkins, R.M.," Design Optimization of a Space Launch Vehicle Using a Genetic Algorithm", JOURNAL OF SPACECRAFT AND ROCKETS, Vol. 45, No. 4, July–August 2008

[13] Schoonover, P.L., Crossley, W.A. and Heister, S.D.," Application of a Genetic Algorithm to the Optimization of Hybrid Rockets", JOURNAL OF SPACECRAFT AND ROCKETS, Vol. 37, No. 5, September–October 2000

[14] Humble, R.W., Henry, Gary. N. and Larson, Wiley, J., 'Space Propulsion Analysis and Design', McGraw Hill, 1995, Space Technology Series
[15] AIAA Standard, Mass Properties Control for Space Systems, 'Mass Properties Control for Space Systems', 2006 AIAA Standard S-120-2006
[16] D.E. Koelle: "Handbook of Cost Engineering for Space Transportation Systems with TRANSCOST 7.0"; Report No. TCS-TR-168/00; TCSTransCostSystems; Ottobrunn, 2000
[17] Isakowitz, S.J., Hopkins, J.B., Hopkins, J.P., International Reference Guide to Space Launch Systems, Fourth Edition, AIAA.
[18] Isakowitz, S.J., Hopkins, J.B., Hopkins, J.P., International Reference Guide to Space Launch Systems, Fourth Edition, AIAA.
[19] US Patent 3,779,009 Joseph Friedman, Catalytic Method of Producing High Temperature Gases.
[20] Mueller, J.M., and McFarlane, J.S., "Design of Tridyne Pressurization System for Liquid Oxygen/Polybutadiene Hybrid Rocket Motors", AIAA/SAE/ASEM 27th Joint Propulsion Conference, AIAA-91-2406, Sacramento, Ca, 1991.
[21] Chakroborty, Dr. Shyama, Wollen, Mark and Malany, Lee, "Development and Optimization of a Tridyne Pressurization System for Pressure Fed Launch Vehicles," 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, AIAA 2006-4716, Sacramento, Ca, 2006.
[22] Markopolous, P., Szedula, J, & Abel, T., "Application of Hybrid Rocket boosters to Launch Vehicle Systems", 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, AIAA 1997-2934, Seattle, WA, 1997.