Advanced Hybrid Rockets for Future Space Launch

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Abstract
Under a contract from the Air Force Research Labs (AFRL), Space Propulsion Group Inc. (SPG) has been developing paraffin-based hybrid rocket technology as a cost-effective, high performance and safe alternative to solid and liquid rockets. SPG's fast burning paraffin-based fuels allow for a single circular port design, eliminating the significant disadvantages of multiport systems, such as the low fuel utilization. In this paper the results of systems studies conducted to replace some of the commonly used solid motors (i.e. GEM60, Castor 30) with advanced hybrid rocket systems has been reported.

1. Introduction

SPG strongly believes that hybrid rocket propulsion is a tipping point technology in the sense that a small, short term investment could have game changing consequences in developing green, safe, affordable and high performance systems needed in future space missions. In order to demonstrate the advantages of hybrids most effectively, the effort should be concentrated on improving the Technology Readiness Level (TRL) of the technology for a carefully selected class of missions. Arguably upper stage motors used in small launch vehicles or strap on boosters for large vehicles constitute the perfect platforms for this purpose. The advanced hybrid rockets that are being developed by SPG are believed to have the capability to deliver high performances required in modern launch vehicles, while retaining the cost, environmental and simplicity advantages of the classical hybrids. In order to demonstrate the performance capabilities of advanced hybrid rockets, a set of design studies have been conducted to replace the Orion 38, Castor 30 and GEM 60 solid rocket motors with LOX/paraffin-based systems.

2. Fundamentals of Commercializing Advanced Hybrid Rockets

Unlike the mature propulsion technologies (such as the solid and liquid rocket systems), which could only result in small incremental improvements, hybrids do have the potential to become a game changing propulsion technology which could potentially lead to significant cost savings in a relatively short period of time. Building on the existing know-how, it is believed that developing advanced paraffin-based hybrid propulsion systems[1 and 2] to be used in a variety of NASA/DoD/commercial missions can be achieved with a relatively low level of effort. Due to the progress already made in the technology development and the inherently cost effective nature of hybrid motor testing, a small investment would lead to great advancements in this field.

SPG has been developing advanced paraffin-based hybrid rocket motors for the last 8 years under a contract from the US Air Force. Over the course the program great advances have been made in achieving a good level of understanding of the critical physical process that take pace in hybrid rocket motors. This understanding has been successfully utilized to build a set of scalable design tools for developing efficient and stable motors for which the inherent simplicity/safety of the hybrid systems could still be retained.

We believe that the first implementation of the advanced paraffin-based hybrid rocket technology should be to a smaller system such as an upper stage motor. Even though this technology is suitable for a wide range applications (including boosters, sounding rockets), the upper stage application mode/market makes more sense for the first implementation due to the following reasons:

- The high specific impulse performance of the liquid oxygen/paraffin-based hybrid rockets gives them a competitive advantage for upper stages which are known to be highly sensitive to this performance parameter.
- Upper stages are typically smaller pressure fed propulsion systems which makes them easier to develop compared to the heavier systems such as boosters used in medium/heavy launch vehicles.
- The technologies developed in this program can be adapted to the systems used in the suborbital space tourism, an emerging market with a large growth potential.

Key properties of some of the existing upper stage propulsion systems are listed in Table 1. Note that the solid rocket motors which are commonly used in small launch systems inherently have low Isp performance. Most LOX/kerosene based liquids operate at non-optimal O/F values, resulting in suboptimal Isp performance for this particular propellant combination. One exception is the Russian RD-58 engine which employs complicated technologies such as oxygen-rich gas generator. Note that despite their excellent Isp performance, LOX/H_2 engines are not ideal for small/medium size motors or air launch systems due to their poor density impulse performance along with the complicated operations associated with the hydrogen fuel. The storable NTO/hydrazine-based liquids have decent Isp performance and impulse density. However the toxicity and environmental unfriendliness of these propellants make the NTO/hydrazine-based liquids highly unpopular in modern systems. It is clear that an affordable and green upper stage system with reasonably good performance would be highly competitive with the systems listed in Table 1.

Table 1: Some of the existing upper stage propulsion systems[3]

	Propellants	**Delivered Isp, sec**	**Launch System(s)**	**Notes**
Orion 38	HTPB	289	Taurus/Pegasus	Low Isp
Orbus 21D	HTPB	293	Athena	Low Isp
Kestrel	LOX/Kerosene	324	Falcon 1 E	Suboptimal
Scorpius Stage 3	LOX/Kerosene	323	Scorpius	Suboptimal
RD-869	UDMH/N2O4	317	VEGA	Toxic, Non US
RD-58	LOX/Kerosene	349	Proton	Non US
RL 10	LOX/H2	450	ATLAS V	Cryogenic

Development of a small upper stage hybrid motor is expected to be completed in a 2-3 year period and would require a relatively low level of investment (compared to the development of a comparable liquid or solid system). The advanced paraffin-based hybrids can also be applied to larger booster systems in the 250-1,000 klbs thrust range. However the higher development costs for these systems makes them a more risky first application for hybrids.

2.1 Advanced LOX/Paraffin-Based Hybrid Rocket Technology

The classical hybrid rocket systems suffer from two major shortcomings: 1) complex multiport fuel grains as a result of the poor regression rate performance of the classical polymeric fuels and 2) low frequency instabilities. In the past, the mitigation methods for these problem areas have introduced significant complexity to the motor design, compromising the simplicity advantage of hybrids. For example, the 250 klb thrust motor developed by American Rocket Company (AMROC) was based on a complex 15 port wagon wheel configuration (resulting in poor fuel utilization and expensive fabrication) and the motor stability was achieved by the continuous injection of a hazardous pyrophoric substance, triethylaluminum (TEA).

SPG’s paraffin-based/LOX hybrid rocket technology, which has an inherently high fuel regression rate, allows for the use of a simple single circular port fuel grain design approach. SPG has also developed a unique proprietary technology to eliminate the low frequency instabilities and acoustic instabilities in LOX-based hybrids without resorting to external heat or pyrophoric liquid addition at the fore end of the motor. These two technological advancements are crucial in keeping the hybrid concept cost effective, simple and safe compared to the state of the art liquid and solid rocket systems. The key virtues of the advanced LOX/paraffin-based motor technology and its impact at the systems level are summarized in Table 2.

Table 2: SPG's enabling hybrid rocket technologies

Virtue	Enabling Key Technology	Impact
Single Circular Port Design	Paraffin-based fuels	Simple, inexpensive grain High fuel utilization
Adjustable fuel regression rate	Fuel formulation tailored to mission	Mission flexibility
Stable and efficient combustion with no external heat or TEA addition	SPG proprietary injector/fore end design	High performance without compromising systems simplicity
High Isp and impulse density	Paraffin-based fuels and LOX	Light and small systems
Low cost, readily available propellants	Paraffin-based fuels and LOX	Reduced development and recurring costs
Simple motor design with no exotic materials	Advanced internal ballistic design and testing	Reduced costs and high reliability
Low environmental impact	Paraffin-based fuels and LOX	Simplified operations and reduced costs
Safety	Zero TNT equivalency Low fire hazard	Ideal for manned systems Reduced development costs
Throttling	SPG proprietary throttling valve developed and tested	Mission flexibility
Efficient gas phase combustion	Tailored fuel grain technology	Effective use of the pressurant as propellant
Thrust vector control (TVC)*	Liquid/Gaseous Injection Thrust Vector Control	Simple TVC capability

* This technology has been developed and demonstrated by AMROC and Whittinghill Aerospace.

Comparison to Other Chemical Systems:

The advanced LOX/paraffin-based hybrids have significant advantages over the solid rocket systems, some of which can be summarized as:

- Cost savings in motor development, manufacturing and launch operations: The savings stem from the inherent safety of the hybrid system and the simplicity of the motor fabrication which can be conducted even in light industrial zones. Transportation is also cost effective due to the non-hazardous classification of the paraffin-based motor. Currently SPG is shipping its motors to the test site using standard freight. Moreover the overall propellant cost for the paraffin-based/LOX system is at least an order of magnitude lower than the cost of a typical solid rocket propellant.
- Inherent safety in manufacturing, transportation, storage, testing and launch operation phases: One of the important implications of the zero TNT equivalency and motor shutdown capability of the advanced hybrid systems is the simplification of the range safety requirements which is known to be a major cost driver for launch systems.
- Higher fault tolerance: Hybrids are not sensitive to cracks or debonding, eliminating the requirement for expensive quality control operations such as x-ray examination of the motor.
- Environmental friendliness: Paraffin, its additives and LOX are all green and non-hazardous materials. Most high performance solid rockets contain ammonium perchlorate, a substance which presents hazard to the environment and shown to have adverse effects on human health[4].
- LOX/paraffin-based hybrids have significant delivered Isp advantage (~35 seconds) over the solid rockets.
- Hybrids can be throttled or can be shut down. This virtue introduces mission flexibility and improves the orbital insertion accuracy.

The primary disadvantage of the hybrid system compared to the solid rocket is its lower impulse density which results in a volume limited design approach for applications with strict geometrical envelope requirements.

Paraffin-based/LOX hybrids have also significant advantages over liquid rockets.

- Mechanical simplicity: Hybrids have significantly simpler liquid storage/feed systems and injectors (one liquid as opposed to two). Moreover no active cooling of the hybrid chamber is necessary, since it is protected by the fuel grain. The simplicity is expected to lead to significant cost savings.
- Structural mass fraction of a well designed LOX/paraffin-based system is expected to be better than a pressure fed LOX/RP-1 liquid rocket. Note that the pump fed liquid rockets are not expected to be cost effective for small propulsion systems.
- Hybrids are fault tolerant compared to liquids. The tolerance requirements on the machined parts can be much more relaxed in hybrids.
- Hybrids have reduced fire hazard compared to liquids.

2.2 State of the Technology

SPG has successfully manufactured and fired numerous LOX/paraffin-based flight weight (carbon composite case) hybrid rocket motors up to 22 inches in diameter. A picture of the 22 inch motor firing conducted in Butte MT is shown in Figure 1. High efficiencies and stable combustion has been demonstrated at different motor scales.

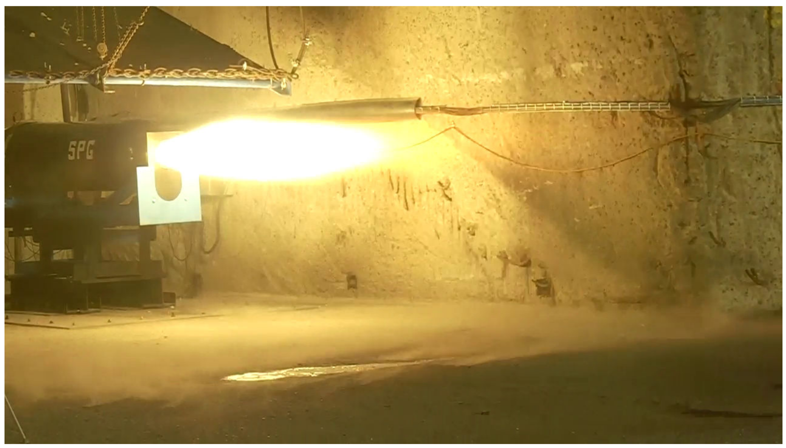


Figure 1: SPG's 22 inch diameter motor firing

3. Systems Studies

In this section, the hybrid replacement systems studies conducted for three modern solid rocket motors will be discussed. The systems selected for this study (Orion 38, Castor 30 and GEM 60) are widely used in the current US based launch vehicles. Orion 38 and Castor 30 are used in upper stages whereas GEM 60 is a strap on booster motor. The characteristic virtue of these solids is their relatively low Isp performance and very high recurring costs.

3.1 Orion 38 Replacement

A systems/optimization study has been conducted to evaluate the performance of a series of paraffin-based hybrid rocket concepts as replacements for the solid rocket motor Orion 38. This motor, which is manufactured by ATK, is selected in this study since

- It is currently used in a wide range of launch vehicles including Pegasus.
- It is a state of the art solid rocket system.

Some of the important properties of the Orion 38 motor are listed in Table 3[5].

In the design process, the total impulses of the hybrid systems have been matched to the Orion 38 value. Total impulse matching is preferred over delta V matching, since the latter requires the arbitrary selection of a payload mass. The burn times for the hybrids have been increased to 120 seconds in order to minimize the component weights and also to reduce the acceleration loading on the payload. Even though a formal optimization on the burn time has not been conducted, it has been determined that the system gross mass was fairly flat within the burn time range of 100-200 seconds. A burn time close to the lower end of this range has been selected to minimize the heat loading on the motor internal components.

Table 3: Summary of data for the Orion 38 solid rocket

Motor	**Orion 38**
Vacuum Total Impulse, N-sec	$2.18\ 10^6$
Burn Time, sec	67.7
Outside Diameter, m (in)	0.965 (38.0)
Average Chamber Pressure, atm (psia)	38.9 (572)
Average Nozzle Area Ratio	49.3
Length, m (in)	1.35 (53.0)
Gross Mass, kg	885.8
Structural Mass Fraction	0.124
Delivered Isp, sec	289.0

The overall diameter for the LOX based systems have been kept below the diameter of the Orion 38 system. For the Nytrox based system this arbitrary assumption has been relaxed. The gross mass and the overall length of the hybrid systems are left free to change to meet the desired performance requirements.

Several propellant combinations have been considered in the study for comparison purposes:

1) LOX/Paraffin: This is the baseline propellant combination using the high performance cryogenic oxidizer. In order to keep the recurring and development costs low, no performance additives have been included in the fuel.
2) LOX/Paraffin-20% AlH_3: In this variation 20% AlH_3 (by mass) in the powder form has been included in the fuel in order to improve the specific impulse and the density impulse performance of the system. Note that the hydrophobic nature of the paraffin binder allows for the safe operation with the water sensitive metal hydrides.
3) Nyrox80/Paraffin-20% Aluminum: This is a storable (days on the launch pad) but lower performance system. Note that the temperature of Nytrox[6] is selected to be -80 C. 20% aluminum powder (by mass) has been included in this variation in order to a) improve the Isp performance, b) densify the fuel and c) increase the regression rate of the fuel. In order to minimize the material and processing costs low, an aluminum powder with micron sized particles has been selected over the nano sized aluminum powder.

Note that only nontoxic, environmentally green, low cost, chemically stable and readily available (with the possible exception of AlH_3) propellants have been considered in this investigation. For simplicity the oxidizer mass flow rate is assumed to be constant in time. Throttling can easily be implemented, if it is required for a certain class of missions.

Due to the relatively small scale of the motors, all hybrids included in this study are selected to be pressure fed systems. It is determined that the details of the pressurization system design on the system performance are critical. Nytrox systems are partially self-pressurized. They start at high pressures but in the blow down mode they lose pressure at a fast rate, since oxygen is expected to come out of solution rather slowly. Thus they require a supplementary pressurization system. This can be achieved either by helium or gaseous oxygen. The advantage of helium is its lightness. However helium is a limited natural resource and it has become quite expensive and inaccessible in recent years. This trend is expected to get worse in the upcoming years. The other alternative is to use gaseous oxygen as the supplementary pressurant for the Nytrox systems. For such a system, at the end of the liquid burn most of the vapor is composed of oxygen (typically more than 90%). Note that the optimal oxidizer to fuel ratio for the oxygen paraffin system is around 2.5 as opposed to 8 in the case of N_2O. As discussed above, the sudden reduction of oxidizer flow rate in transition to the vapor phase, which results in a significant drop in the oxidizer to fuel ratio, is ideal for an oxygen/paraffin hybrid. It is shown by motor tests that the combustion process in this GOX hybrid operating close to its optimum O/F can be sustained much more readily than the combustion with the pure N_2O vapor (also note that oxygen is a much more energetic oxidizer). SPG's extensive experience with gaseous oxygen/paraffin hybrids indicates that the oxygen rich gas phase combustion mode is expected to be stable and efficient. Based on these arguments, we have included the contribution of the gas phase combustion to the total impulse. The high efficiency gas phase combustion in oxygen pressurized Nytrox systems has been demonstrated in motor testing.

Gaseous oxygen is selected as the pressurant gas also for the LOX motors. Note that combustion of the gaseous pressurant in the hybrid motor chamber is also feasible in the LOX based systems.

The systems calculations have been conducted using SPG's hybrid vehicle design code. Since the complicated mission requirements have been replaced by the simple total impulse matching on the propulsion system, the flight modules have been disabled. The structural mass fraction is estimated within the code using the preliminary design equations for the major components such as the oxidizer tank, combustion chamber, pressurization system and nozzle. Available mass data have been used for small components such as valves, regulators and ignition system. A mass margin of 20% has been added to the calculated structural mass. The code outputs all of the relevant performance, weight and geometrical parameters which are summarized in Tables 4, 5 and 6 for the systems considered in this study. For simplicity, the cost function for the optimization process is taken to be the gross mass of the motor. Note that for an upper stage system any reduction in the gross mass corresponds to an increase in the payload mass for a given delta V requirement.

The oxidizer tank material selected for the Nytrox system is aluminum lined carbon-epoxy. For the LOX systems three tank materials have been considered in this investigation: 1) Low cost option: Al 2219, 2) Al 2195- Weldalite and 3) linerless composite. The combustion chambers are made out of graphite-epoxy composite with a thermoset liner.

For the cryogenic LOX system, an insulation thickness of 0.5 inches has been used in the calculations. For the Nytrox system the insulation thickness of 0.2 inches has been determined to be adequate even for long storage periods on the launch pad.

In order to simplify the fabrication process and minimize the fuel sliver fraction, all motors are designed to use a single circular port fuel grain. The initial mass flux selection is based on the limitations on the port Mach number and the hoop stresses that occur on the port surface due to internal pressure loading. The thickness of the motor insulation liner is 0.2 inches for all systems. A fuel sliver thickness of 0.1 inches has been assumed at the end of the burn. This corresponds to a fuel utilization of 98% or better for the systems considered in this study. Note that the small sliver fractions have been successfully demonstrated in paraffin-based hybrid motor testing.

The nozzles are made out of ablative silica phenolic inner shell and a structural outer shell made out of glass phenolic. An erosion rate 0.005 in/sec has been assumed at the reference chamber pressure of 300 psi. A linear variation for the erosion rate is assumed with the chamber pressure. Moderate variations in the erosion rate are not expected to affect the system performance appreciably.

For systems with no performance additives, the c* and nozzle efficiencies are assumed to be 0.96 and 0.98, respectively. For the systems utilizing fuel grains with Al or AlH_3 additives, the nozzle efficiency is reduced to 0.97 to account for two phase losses. Note that the assumed combustion efficiencies have been demonstrated in motor testing conducted by SPG. An injector pressure drop of 70 psi is assumed for all motors.

Another key advantage of the hybrids over solid systems is related to the fact that, in a hybrid motor, the chamber pressure can be used as a free parameter in the optimization process due to the lack of pressure dependency of the regression rate. Since chamber pressure can be selected independent of the internal ballistic considerations, more effective optimization is possible (essentially one of the constraints is lifted). The other parameters included in the optimization process are O/F and nozzle expansion ratio.

In the baseline configuration, as shown in Figure 2, a total of 8 cylindrical tanks have been used parallel to the combustion chamber. This particular scheme is selected to minimize the overall system length and diameter of the motor. A total of 4 spherical pressurization tanks have been used in the baseline design. The liquid oxidizer is fed into the motor by the use of dip tubes in each tank. Note that dip tubes are commonly used in the ground testing of hybrid rocket motors.

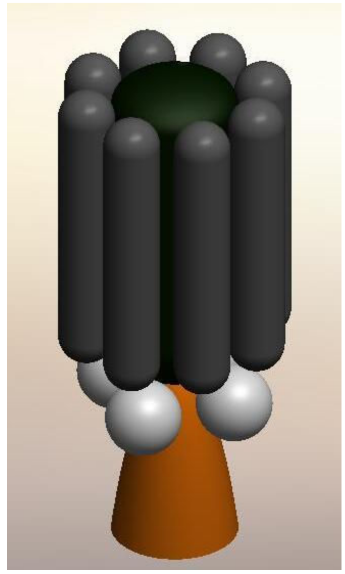


Figure 2: Baseline motor configuration: Cylindrical tanks in parallel with the chamber

A variant of the baseline design uses one torroidal oxidizer tank as opposed to 8 cylindrical tanks (Figure 3a). Even though the torroidal tanks are heavier and more expensive than the cylindrical tanks, for missions requiring compactness this design could be beneficial.

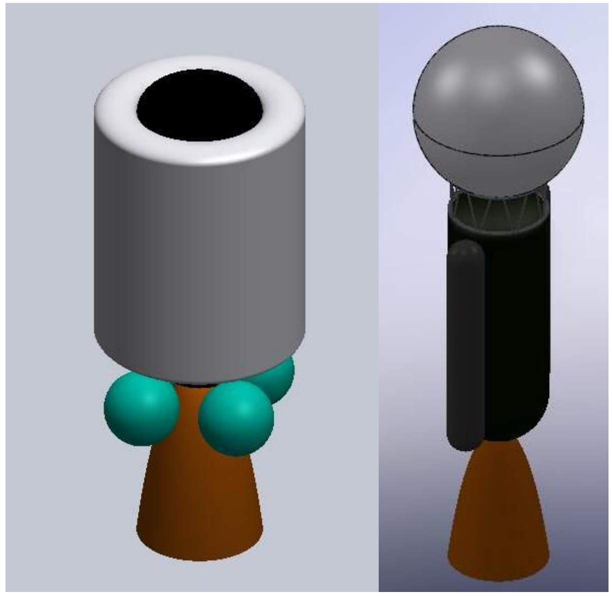
Finally, if the length is not a critical constraint, a system that uses a single spherical tank in series with the combustion chamber could be used (Figure 3b). This long configuration is the most efficient structural design due to the use of a single spherical tank for oxidizer storage.

The general properties of the optimum Orion-38 replacement motors are summarized in Table 4. As shown in Table 4, despite their reduced chamber pressures, all systems, deliver a significantly better Isp than the Orion-38 system. The Isp for the LOX motors are significantly higher. The structural mass fractions (stage burn out mass over the stage gross mass) of the replacement systems are in the 0.120-0.139 range. This range covers the reported Orion-38 value. Note that hybrids optimize at much lower chamber pressure levels (200 psia). As expected, the systems using tanks made out of Weldalite and linerless carbon composite have lower structural mass fractions.

The propulsion system weights for all systems are listed in Table 4 for the Orion-38 replacement. Note that all hybrids are distinctly lighter than the Orion-38 motor. Specifically the LOX based systems are 15.2-18.1% lighter and the Nytrox based system is 8.9% lighter. The weight distribution for the major components is shown in Table 5. The “other mass” category reported in Table 5 includes small components such as valves, regulators, piping and structures.

The overall lengths and diameters of the hybrid systems are also included in Table 4. Note that all systems are longer than the solid motor due to the low effective density of the hybrid propellants. The shortest of all the systems considered in this study is the aluminized Nytrox80/paraffin motor. All LOX based systems are slightly longer than the Nytrox option. In this study no effort has been made to minimize the length of the replacement hybrid systems. We believe that substantial reduction in length is feasible with small penalty in the vehicle gross mass (the gross mass is very flat around the optimum point).

In order to quantify the payload increase caused by switching to high performance hybrid motors for the upper stage, a launch vehicle resembling the Pegasus system is used as an example. Orion 50S and Orion 50 motors are used as the first and second stages, respectively. For the baseline system, Orion 38 has been used as the upper stage motor. For the all solid launch vehicle, the payload capability has been estimated as a function of mission delta V and plotted in Figure 4. Similar calculations have been conducted using the hybrid replacements instead of the Orion 38. For simplicity we have limited the study to two hybrid motors: 1) baseline LOX/paraffin and 2) high performance LOX/paraffin-20% AlH_3. The first two stages are kept the same for all calculations.



(a) (b)

Figure 3: Optional motor configurations: a) Torroidal tank, and c) Spherical tank in series

The payload mass is plotted as a function of mission delta V in Figure 4 for launchers using the hybrid upper stage. As shown in the figure the payload capability for the hybrids are significantly higher than the solid based system. The high performance hybrid system delivers a slightly more payload compared to the baseline LOX/paraffin motor. Figure 5 contains a plot of the percent payload mass increase over the Orion-38 solid motor as a function of mission delta V. Note that substantial payload increases has been achieved especially for demanding missions requiring large delta V performance (i.e. 40% payload increase at a mission delta V of 9,500 m/sec).

Table 4: Properties of the systems considered in the systems study

	Baseline	**Weldelite Tank**	**Composite Tank**	**High Performance**	**Storable**
Propellant	LOX/Paraffin	LOX/Paraffin	LOX/Paraffin	LOX/Paraffin-20% AlH3	Nytrox80/ Paraffin-20% Al
Oxidizer Tank Material	Al 2219	Al 2195-Weldalite	Linerless Composite	Al 2195-Weldalite	Al Lined Composite
Average O/F	2.7	2.7	2.7	2.2	6.0
Chamber Pressure, psia	200	200	200	200	200
Initial Nozzle Area Ratio	60	60	60	60	70
Burn Time, sec	120	120	120	120	120
Effective Isp, sec	338.0	338.0	338.0	343.1	310.1
Overall Length, in	106.6	106.6	106.6	107.3	82.8
Overall Diameter, in	37.2	37.1	37.1	35.6	45.5
Structural Mass Fraction	0.139	0.126	0.120	0.121	0.126
Gross Mass, kg	751.2	740.4	735.4	725.1	807.0
Gross Mass Decrease, %	**15.2**	**16.4**	**17.0**	**18.1**	**8.9**

Table 5: Mass breakdown for the hybrid systems considered in the study

System	Baseline	Weldelite Tank	Composite Tank	High Performance	Storable
Propulsion System Mass, lb	1,668	1,644	1,633	1,610	1,792
Structural Mass, lb	231.9	207.5	196.6	194.9	226.6
Tank Mass, lb	67.1	47.2	38.4	37.0	74.9
Chamber Mass, lb	46.1	46.1	46.1	46.7	28.2
Pressurization System Mass*, lb	17.2	17.2	17.2	16.3	20.1
Nozzle Mass, lb	32.6	32.6	32.6	32.6	37.3
Other Mass, lb	29.5	28.9	28.6	28.9	27.1
Margin, lb	39.5	35.5	33.7	33.3	39.0

*Excluding burned gas mass

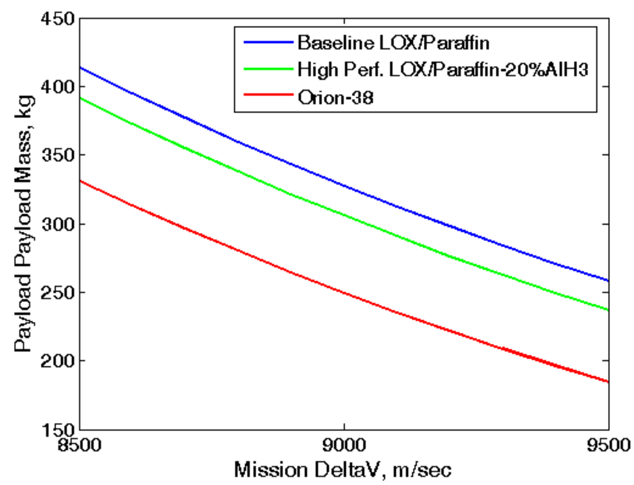


Figure 4: Payload mass as a function of mission delta V for three different upper stage motors

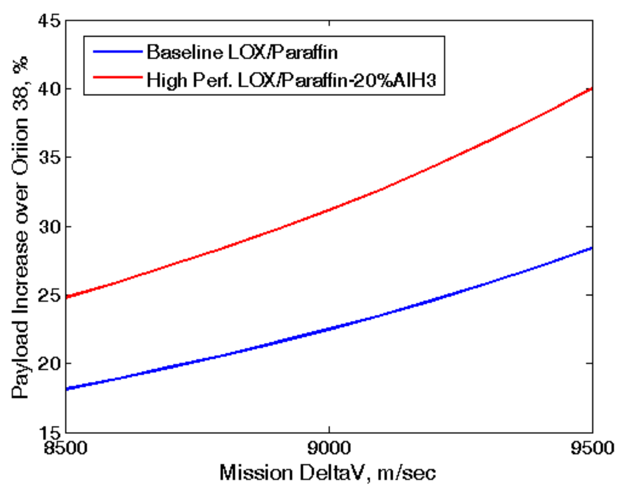


Figure 5: Payload mass increase over Orion-38 as a function of mission delta V

3.2 Castor 30 Replacement

A system study has been conducted to evaluate the performance of a LOX based advanced hybrid rocket concept as replacement for the Castor 30 solid rocket motor. Castor system is currently being used as the upper stage for some of the modern US launchers such as Taurus. A high performance, low cost replacement for this expensive solid motor is expected to improve the cost per payload performance of the launch vehicle. Properties of the Castor 30 system are summarized in Table 6[5].

Table 6: Summary of data for Castor 30 solid motor

Motor	**Castor 30**
Vacuum Total Impulse, N-sec	37.0 10^6
Burn Time, sec	143
Outside Diameter, m (in)	2.34 (92)
Length, m (in)*	5.26 (207)
Nozzle Area Ratio	50.0
Chamber Pressure, psi (bar)	762 (51.8)
Gross Mass, kg	13,960
Structural Mass Fraction	0.087
Vacuum Isp, sec	294.7

* Estimated from nozzle length

The configuration selected for the Castor 30 system is shown in Figure 6. Four cylindrical oxidizer tanks are used in parallel configuration in order to minimize the length of the stage. The outside diameter is limited to the diameter of the Taurus launch vehicle's third stage diameter. The pressurization tanks are integrated to the oxidizer tanks (they use a common bulk head design). The initial value of the pressurization system pressure is selected to be 3,000 psi.

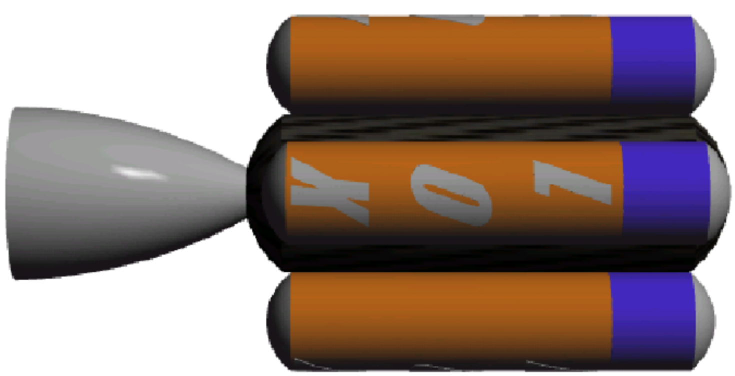


Figure 6: Configuration for the Castor 30 replacement system

The following material selections have been made in the design:

- Motor: Carbon composite with glass composite liner
- Tank: Al 2219 is used for affordability
- Pressurization: Al 2219 since common bulkhead design has been implemented
- Nozzle: Silica phenolic ablator with glass phenolic shell.

The c* and nozzle efficiencies are assumed to be 0.96 and 0.98, respectively. Nozzle erosion rate is 0.005 in/sec at the reference chamber pressure of 300 psi. A mass margin of 25% has been added to the estimated burn out mass of the stage. The properties of the Castor 30 replacement are listed in Tables 7-9. Note that the hybrid is 9% lighter than the solid which is expected to improve the payload performance of the launch vehicle by 15-20%.

As shown in Table 9, hybrid is only 13% longer than the solid.

Table 7: Summary of the performance parameters for the hybrid systems considered in the study (Castor 30).

System	LOX/Paraffin
Gas Phase Impulse, %*	6.62
Oxidizer Temperature, C	-183
Oxidizer Tank Material	Al. 2219
Oxidizer Specific Density	1.140
Pressurization Scheme	GOX-pressurized
Oxidizer to Fuel Ratio	2.7
Chamber Pressure, atm (psi)	20.4 (300)
Effective Isp, sec	334.3
Structural Mass Fraction	0.110
Propulsion System Mass***, kg	12,712
Increase in Mass, %	-9.0

* Percent of total impulse, ** Percent of gross mass, ***Castor 30 gross mass is 13,960 kg

Table 8: Summary of the component masses for the hybrid systems considered in the study. (Castor 30)

System	LOX/Paraffin
Propulsion System Mass, kg	12,712
Structural Mass, kg	1,398*
Tank Mass, kg	299
Chamber Mass, kg	387
Pressurization System Mass**, kg	209
Nozzle Mass, kg	103

*Including 25% mass margin **Excluding gas mass

3.3 GEM 60 Replacement

A system study has been conducted to evaluate the performance of a LOX based advanced hybrid rocket concept as replacement for the GEM 60 solid rocket motor. GEM 60 system is currently being used as the strap on boosters for the Delta Medium launch vehicle. A low cost replacement for this expensive solid motor is expected to make the Delta vehicle more competitive in the market and save money for its customers. Properties of GEM 60 system are summarized in Table 10[5].

Table 9: Summary of the dimensions for the hybrid systems considered in the study. (Castor 30)

System	LOX/Paraffin
Initial Nozzle Area Ratio	40:1
Effective Nozzle Area Ratio	36.7:1
Tank Diameter, m (in)	0.864 (34.0)
Tank Length, M (in)	3.198 (125.9)
Motor Diameter, m (in)	1.377 (54.2)
Motor Length, M (in)	4.234 (166.6)
Pres System Diam, m (in)	0.864 (34.0)
Pres System Length, m (in)	1.143 (45.0)
Nozzle Exit Diameter, m (in)	1.509 (59.4)
Nozzle Length, m (in)	2.017 (79.4)
Overall Length, m (in)	5.951 (234.3)
Overall Diameter, m (in)	3.101 (122.6)
Length Increase*, %	13.2

* Length of the Castor 30 system (including nozzle) is estimated to be 207 inches.

Table 10: Summary of data for GEM 60 solid motor

Motor	GEM 60
Vacuum Total Impulse, N-sec	76.4 10^6
Burn Time, sec	89.7
Outside Diameter, m (in)	1.52 (60)
Length, m (in)	16.2 (636)
Nozzle Area Ratio	11.0
Chamber Pressure, psi (bar)	1,312 (90.5)
Gross Mass, kg	33,199
Structural Mass Fraction	0.10
Vacuum Isp, sec	275.2

In the design process the following parameters are matched to their values for the solid motor:

- Total impulse
- Burn time
- Effective area ratio
- Outside diameter

The effective nozzle area ratios are matched in order to obtain a direct comparison of the Isp performance with the solid motor. The gross mass and the overall length of the hybrid systems are left free to change to meet the desired performance requirements.
Only the LOX/paraffin propellant combination has been included in this study. Fuel is straight paraffin, since aluminum addition does not increase the specific impulse performance for high energy oxidizers such as oxygen.

The LOX-based system is the other extreme for which all of the pressurization is supplied by a foreign gas. In this case helium is used to minimize the weight of the pressurant gas. Helium constitutes a very small fraction of the overall system mass due to its light nature.

The systems calculations have been conducted using SPG's hybrid vehicle design code. Since the complicated mission requirements have been replaced by the simple total impulse requirement on the propulsion system, the flight modules have been disabled. The structural mass fraction is estimated within the code using the preliminary design equations for the major components such as the oxidizer tank, combustion chamber, pressurization system and nozzle. Available mass data have been used for small components such as valves, regulators and ignition system. A mass margin of 20% has been added to the calculated structural mass. The code outputs all the relevant performance, weight and geometrical parameters which are summarized in Tables 11-13 for all systems considered in this study.

The oxidizer tank material for the LOX-based system is selected to be aluminum 2219. The combustion chambers are also made out of graphite-epoxy composite with glass-epoxy liner. For the cryogenic LOX system, an insulation thickness of 2.54 cm (1.00 inch) has been used. Note that the outside diameter of the insulation was matched to the diameter of the GEM 60 system.

In order to simplify the construction process and minimize the fuel sliver fraction, the motor is designed to use a single circular port fuel grain. The initial to final diameter ratio for the fuel port is taken to be 2.0 to limit the port Mach numbers and the hoop stresses that occur on the port surface due to internal pressure loading. The thickness of the motor insulation liner is 0.64 cm (0.25 inches). A fuel sliver thickness of 0.25 cm (0.1 inches) has been assumed at the end of the burn (liquid and gas when possible). This corresponds to fuel utilization of 97% or better for the systems considered in this study. These small slivers have been successfully demonstrated in paraffin-based hybrid motor testing. The nozzles are made out of ablative silica phenolic inner shell and a structural outer shell made out of glass phenolic. The nozzle erosion rates are extrapolated from actual motor test data. An erosion rate of 0.1524 mm/sec (0.006 in/sec) has been assumed at the reference chamber pressure of 34.0 atm (500 psi). A linear variation for the erosion rate is assumed with the chamber pressure. Moderate variations in the erosion rate are not expected to affect the system performance significantly. For all systems the c* and nozzle efficiencies are assumed to be 0.96 and 0.98, respectively. Note that the combustion efficiency value has been demonstrated in motor tests conducted by SPG.

Even though the chamber pressures (peak values) for the solid motor is 90.5 atm (1,312 psi), the hybrid system has optimized at a lower average pressure of 40.8 (600 psi). Note that this is due to the pressure independent regression rate behavior, which is another key advantage of hybrids compared to solids. Since chamber pressure can be selected independent of the internal ballistic considerations, more effective optimization is possible (essentially one of the constraints is lifted).

Table 11: Summary of the performance parameters for the hybrid systems considered in the study (GEM60).

System	**LOX/Paraffin**
Gas Phase Impulse, %*	0.0
Oxidizer Temperature, C	-183
Oxidizer Tank Material	Al. 2219
Oxidizer Specific Density	1.140
Pressurization Scheme	He-pressurized
Oxidizer to Fuel Ratio	2.7
Chamber Pressure, atm (psi)	40.8 (600)
Effective Isp, sec	306.2
Structural Mass Fraction	0.1921
Propulsion System Mass***, kg	31,548
Increase in Mass, %	-5.0

* Percent of total impulse, ** Percent of gross mass, ***GEM 60 gross mass is 33,199 kg

The results are summarized in Tables 11-13 for the Gem 60 replacement system. As shown in Table 11, despite the reduced chamber pressures, the hybrid system can deliver a much better Isp than the GEM60 motor. The structural mass fraction of hybrid system is high (around 19%) due to the use of the low cost aluminum tank material.

The propulsion system weight for the replacement is listed in Table 11. Note that despite the poor mass fraction, the hybrid system is almost 5% lighter than the solid.

The weight distribution for the major system components is shown in Table 12. The important observation is the high tank weight for the oxidizer.

Table 12: Summary of the component masses for the hybrid systems considered in the study. (GEM 60)

System	LOX/Paraffin
Propulsion System Mass, kg	31,548
Structural Mass, kg	6,062
Tank Mass, kg	3,431
Chamber Mass, kg	889
Pressurization System Mass*, kg	215
Nozzle Mass, kg	400

*Excluding gas mass

Table 13: Summary of the dimensions for the hybrid systems considered in the study. (GEM 60)

System	LOX/Paraffin
Initial Nozzle Area Ratio	12.2
Effective Nozzle Area Ratio	11.1
Tank Diameter, m (in)	1.47 (50.0)
Motor Diameter, m (in)	1.00 (52.8)
Overall Length, m (in)	23.40 (921.2)
Length Increase*, %	44.8

* Length of the GEM 60 system is 16.2 m (636 inches).

CAD model of the GEM 60 replacement integrated to the Delta Medium Launch Vehicle is shown in Figure 7.

Hybrid is 44% longer than the solid, primarily due to the lower density of the propellants. This length increase is not expected to generate any major systems issues in the Delta Medium application due to the very long nature of the core vehicle (see Figure 7).

4. Conclusions

The following general conclusions can be drawn:

- Hybrid rockets are a tipping point technology. A small investment could make a big difference in the field of chemical rocket propulsion. In the case of matured solid and liquid technologies, improvements are expected to be gradual.
- A well designed hybrid can deliver good performance while retaining its inherent simplicity, safety and cost advantages.

- It would be beneficial to develop a high performance system to demonstrate the capability of advanced hybrids. An upper stage motor is an ideal application.
- The advanced hybrid rockets designed as replacements for the Orion 38, Castor 50 and GEM 60 solid rocket motors are appreciably lighter than the solids.
- A simple launch vehicle calculation revealed that the use of hybrid upper stages instead of the Orion 38 motor could result in substantial increases in the payload capability. Improvements of 40% or higher are possible for demanding missions.
- Based on SPG's existing experience base (in manufacturing motors up to 22 inches in diameter), hybrids are expected to be substantially less expensive than the solid systems in the same impulse class.

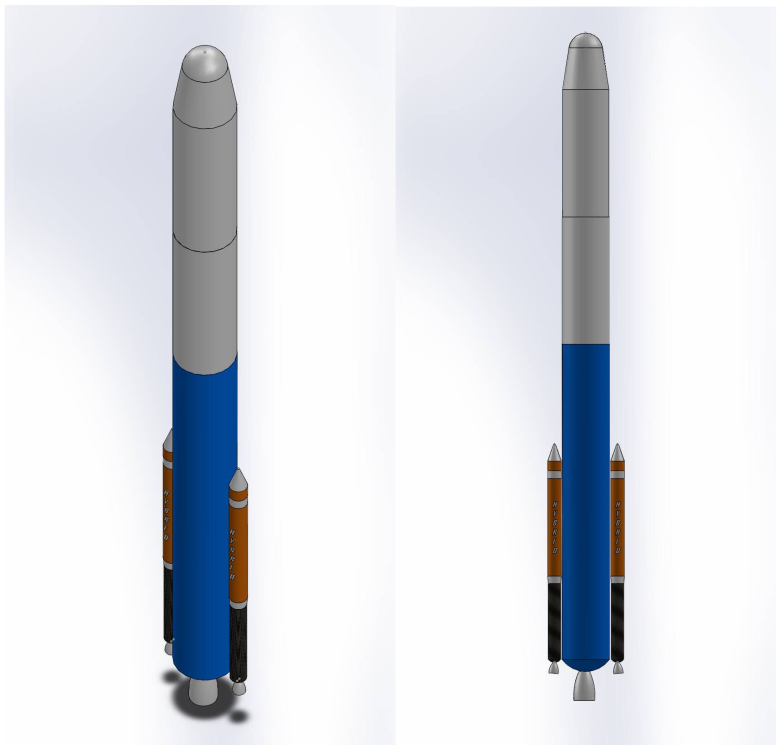


Figure 7: GEM 60 replacement hybrids integrated to the Delta Medium launch vehicle

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