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RECENT ADVANCES IN HYBRID PROPULSION[1]

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The idea of the hybrid rocket is to store the oxidizer as a liquid and the fuel as a solid, producing a design that minimizes the chance of a chemical explosion. While the hybrid enjoys many safety and environmental advantages over conventional systems, large hybrids have not been commercially viable. The reason is that traditional systems use polymeric fuels that evaporate too slowly, making it difficult to produce the high thrust needed for most applications. Research at Stanford University and Space Propulsion Group (SPG) has led to the development of paraffin-based fuels that burn at regression rates 3–4 times that of polymeric fuels. Under the action of the oxidizer flow, the new fuels form a thin, hydrodynamically unstable liquid layer on the melting surface of the fuel. Entrainment of droplets from the liquid–gas interface can substantially increase the rate of fuel mass transfer, leading to a much higher surface regression rate than can be achieved with a conventional fuel. To demonstrate the use of these fuels, a series of scale-up tests using several oxidizers has been carried out on intermediate-scale motors. The data from these tests are in agreement with small-scale, low-pressure, and low-mass-flux laboratory tests and confirm the high regression rate behavior of the fuels at chamber pressures and mass fluxes representative of commercial applications. Recently, SPG has developed a new class of oxidizers based on refrigerated mixtures of N_2O and oxygen. The mixtures combine the high vapor pressure of dissolved oxygen with the high density of refrigerated N_2O to produce a self-pressurizing oxidizer with high density and good performance. The combination of these technologies leads to a hybrid rocket design with reduced system size and mass.

KEY WORDS: *rocket, hybrid, propulsion, paraffin, fuel, oxidizer, combustion, entrainment*

1. ADVANTAGES OF HYBRID ROCKETS

A high regression rate is a natural attribute of paraffin-based fuels, avoiding the need for oxidizing additives or other regression rate enhancement schemes that often compromise safety and drive up cost. The fuels provide specific impulse (I_{sp}) performance comparable to kerosene but are approximately 17% more dense than kerosene. This permits the design of a high volumetric loading single-port hybrid rocket system with a density impulse comparable to or greater than a hydrocarbon fueled liquid system. The high regression rate hybrid removes the need for a complex multiport grain and most applications up to large boosters can be designed with a single-port configuration. To

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further demonstrate the feasibility of this approach, a series of scale-up tests using several oxidizers including gaseous oxygen (GOx), liquid oxygen (LOx), and nitrous oxide (N_2O) has been carried out on intermediate-scale motors. The data from these tests are in agreement with small-scale, low-pressure, and low-mass-flux laboratory tests and confirm the high regression rate behavior of the fuels at chamber pressures and mass fluxes representative of commercial applications. Recently, Space Propulsion Group (SPG) engineers have developed a new class of safe, nontoxic, noncryogenic, self-pressurizing oxidizers based on refrigerated mixtures of nitrous oxide and oxygen called Nytrox. The mixtures combine the high vapor pressure of dissolved oxygen with the high density of refrigerated nitrous oxide to produce a safe, nontoxic, self-pressurizing oxidizer with high density and good I_{sp} performance. These features substantially reduce the overall system size and weight of a hybrid rocket design.

The hazardous operation of the two basic types of chemical rocket propulsion comes mainly from the oxidizer and fuel that must be mixed to release energy in the rocket combustion chamber. In liquid bipropellant rockets, a pump leak or tank rupture that brings these chemicals together in an uncontrolled way can result in a large explosion. In solid propellant rockets, the fuel and oxidizer are already mixed and held together in a polymer binder. Cracks or imperfections in the propellant can cause uncontrolled combustion and explosion.

The idea of a hybrid rocket is to store the oxidizer as a liquid and the fuel as a solid, producing a design that is less susceptible to chemical explosion than conventional solid and bipropellant liquid designs. The fuel is contained within the rocket combustion chamber in the form of a cylinder, with one or more channels called ports hollowed out along its axis as shown in Fig. 1, illustrating a single-port design.

Chiaverini and Kuo (2007) provide an up-to-date, comprehensive discussion of the fundamentals of hybrid rocket operation. Combustion takes place through diffusive mixing between vaporized oxidizer flowing through the port and fuel evaporating from the solid surface. The idealized sketch in Fig. 2 illustrates the flow configuration. The flame thickness and location in the boundary layer are shown roughly to scale. The flame zone is relatively deep in the boundary layer and, according to Marxman and Gilbert (1963) and Marxman et al. (1964), the flame tends to be fuel-rich based on low flame temperatures measured in the boundary layer and on the measured flame position in the boundary layer. Chapter 2 of Chiaverini and Kuo (2007), written by Chiaverini, provides a detailed description of the complex physics of hybrid combustion.

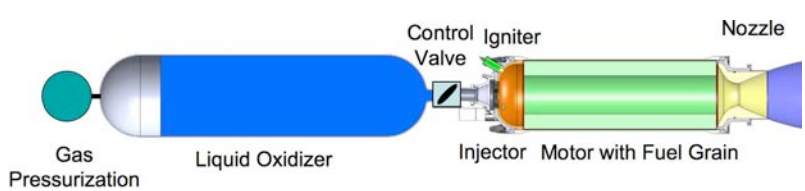

FIG. 1: Single-port hybrid rocket cross-section.

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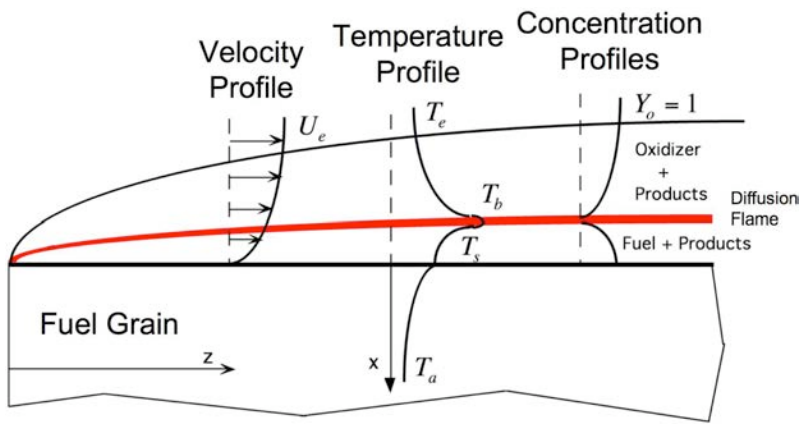

FIG. 2: Boundary layer combustion mechanism for hybrid rockets.

The fuel surface regression rate, $\dot{r}$, where r is the radius of the port, is determined by the oxidizer mass flux in the port and generally follows the theory developed by Marxman and Gilbert (1963) and Marxman (1967) in the early 1960s. The regression rate law is most simply expressed in the form

$$\dot{r} = \alpha \left(\frac{\dot{m}_0}{A_{port}} \right)^n = \alpha G_o^n \tag{1}$$

where $\dot{m}_0$ is the oxidizer mass flow rate, A_{port} is the port cross-sectional area, and $G_o = \dot{m}_o / A_{port}$ is the oxidizer mass flux in the port. The regression rate is primarily governed by turbulent mixing and heat transfer in the boundary layer. In marked contrast to solid rockets, the regression rate of a hybrid is insensitive to the chamber pressure except at very low fluxes where radiation effects become important and at very high fluxes where chemical kinetics effects are important. In a solid, the flame is much closer to the fuel surface and the regression rate is typically an order of magnitude larger.

The parameters α and n are empirical constants for a given set of propellants. The exponent n is typically between 0.5 and 0.8. The remarkable set of data measured by Wooldridge and Muzzy (1965) on wall injection with and without combustion generally supports Marxman's model. Paul et al. (1982) modified Marxman's theory in an attempt to more accurately include the molecular weight variation across the boundary layer due to pyrolysis of high molecular weight polymeric fuels such as hydroxyl-terminated polybutadiene (HTPB).

Theory suggests that the parameter α has a weak inverse dependence on the length of the fuel grain, $\alpha \sim x^{-0.2}$. This would suggest that the regression rate law should exhibit some dependence on the physical size of the motor. However, in the experiments we have conducted to date on a fairly wide range of motor sizes, we have detected little or no length effect for paraffin-based fuels (Karabeyoglu et al., 2005a,b).

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1.1 Comparison to Solid Rockets

While the hybrid rocket concept has been known for over 75 years, it was not given serious attention until the 1960s. See the reviews by Mukunda et al. (1979) and Altman (1991). The primary reason for interest in the hybrid was the nonexplosive nature of the design, which led to safety in both operation and manufacture. The fuel could be fabricated at any conventional commercial site and even at the launch complex with no danger of explosion. Thus, a large cost saving could be realized both in manufacture and launch operation.

Additional advantages of the hybrid over the solid rocket are greatly reduced sensitivity to cracks and debonds in the propellant, higher specific impulse, throttle-ability to optimize the trajectory during atmospheric launch and orbit injection, and the ability to thrust terminate/restart on demand. The products of combustion are environmentally benign compared with conventional solids that generally use perchlorate-based oxidizers. Solid rocket combustion products contain acid-forming gases such as hydrogen chloride. In addition, there are concerns about the effects of low levels of environmental perchlorate, most of which comes from propellant and explosive manufacture. Blount et al. (2006) document the correlation between very low levels of perchlorate contamination and thyroid hormone levels in a large sample of men and women. Women with normally low hormone levels are found to be especially sensitive.

1.2 Comparison to Liquid Bipropellant Rockets

The main advantages of the hybrid over a liquid bipropellant system are as follows:

1. The hybrid rocket presents a reduced explosion hazard compared to a liquid, because an intimate mixture of oxidizer and fuel is not possible.

2. The hybrid rocket requires one rather than two liquid containment and delivery systems. The complexity is further reduced by omission of a regenerative cooling system for both the chamber and nozzle.

3. Throttling control is simpler because it alleviates the requirement to match the momenta of the dual propellant streams during the throttling operation. Throttle ratios up to 10 have been achieved with relative ease in hybrid rocket motors.

4. The theoretical specific impulse of a hybrid rocket is more appropriately compared to a bipropellant liquid than a solid. This is because the oxidizers are the same and the solid fuels are hydrocarbons with energy content similar to kerosene. However, hybrid solid fuel densities are typically 15–20% greater than the density of liquid kerosene. Figure 3 depicts the specific impulse of LOx burning with paraffin and HTPB.

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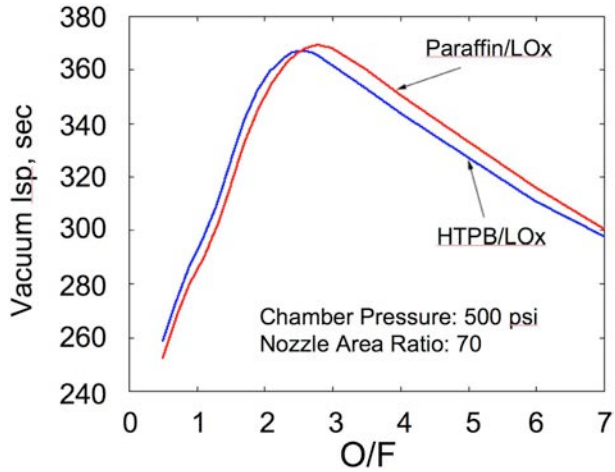

FIG. 3: Vacuum specific impulse vs O/F for Paraffin and HTPB burning with LOx.

5. The fact that the fuel is in the solid phase makes it very easy to add performance-modifying materials such as aluminum powder. The heat of reaction of aluminum is substantial enough to outweigh the increase in molecular weight of the exhaust products, enabling the hybrid to gain a small increase in I_{sp} depending on the oxidizer. The main benefit of the aluminum addition is a substantial increase in fuel density over a comparable hydrocarbon fueled liquid system. There is also a somewhat subtle effect that occurs with storable oxidizers such as N_2O_4, wherein aluminum addition to the fuel tends to both increase the theoretical I_{sp}, and shift the peak I_{sp} to lower values of the oxidizer to fuel (O/F) ratio. For the same propellant total mass there is a larger proportion of the more dense solid propellant. This leads to a reduced liquid feed system and tank size, producing better performance. This point is addressed further in the discussion related to Fig. 10.

6. The ability to add energetic materials to the fuel is one of the key advantages of hybrids over bipropellant liquids. Calabro (2004) suggested the addition of aluminum trihydride (AlH_3, called alane) to HTPB in a LOx hybrid design as a possible replacement for the Ariane 5 boosters. Calculations by Calabro suggest a possible increase in the ideal specific impulse by as much as 25 s compared to a LOX/HTPB hybrid. Approximately the same level of performance increase would occur with the addition of alane to paraffin fuel. DeLuca et al. (2009) provide an excellent discussion of the preparation and handling of the energetic but hazardous alane along with a characterization of its physical properties. They note that the α form of AlH_3 is quite stable at room temperature and therefore may be useful as a propellant additive for both solid and hybrid rockets. They show a significant increase in the regression rate of both HTPB and paraffin at low oxidizer

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flux probably associated with ejection of fuel into the port due to hydrogen outgassing. Alane (1.48 g/cm^3) also increases fuel density, although not as much as pure aluminum (2.70 g/cm^3). There are additional benefits from metal hydrides; Calabro et al. (2007) show that adding magnesium hydride to the fuel substantially reduces the concentration of oxidizing species in the nozzle flow, an effect that would reduce nozzle erosion and further improve performance.

1.3 The O/F Ratio

Over the course of a burn at a fixed oxidizer mass flow rate, there is a tendency for the O/F ratio to shift to higher values as the port opens up. This can be seen from the following. For a single circular port:

$$O/F = \frac{\dot{m}_0}{\dot{m}_f} = \frac{\dot{m}_0}{\rho_f A_b \dot{r}} = \frac{\dot{m}_0}{\rho_f L \pi D \alpha \left(\frac{4\dot{m}_0}{\pi D^2}\right)^n} = \frac{\dot{m}_0^{1-n} D^{2n-1}}{4^n \pi^{1-n} \alpha \rho_f L} \tag{2}$$

where A_b is the port surface area, L is the port length, D is the port diameter, and ρ_f is the fuel density. Recall that the exponent is generally in the range $0.5 < n < 0.8$. As the port diameter increases, the burning area increases and the oxidizer mass flux goes down. For $n > 0.5$, the decrease in mass flux dominates the increase in burning area and the overall fuel mass flow rate goes down. The net effect is to cause the chamber pressure and hence the thrust to decrease naturally over the course of the burn as the vehicle mass decreases. This feature is desirable for a launch system where the payload is subject to a maximum acceleration constraint. Compare this with a solid rocket where the propellant regression rate increases with chamber pressure and the thrust tends to increase during the burn. Furthermore, in a solid rocket a throttling option is generally not available, requiring a complex port design to prevent excessive force on the payload.

Note that the O/F shift implied by Eq. (2) means that the specific impulse cannot be maintained at its peak value at constant oxidizer mass flow rate. Figure 3 shows that for oxygen burning with paraffin or HTPB ($n > 0.5$), the performance can drop off quite steeply as the O/F moves away from the value corresponding to peak I_{sp}. The change of O/F ratio with the opening up of the port implies a change in specific impulse and a possible reduction in vehicle performance. This is a factor that must be taken into account by the designer seeking to get maximum total delivered impulse from the motor. In reality, the exponent $2n - 1$ is quite small and, if one designs the hybrid to begin burning at an O/F somewhat to the left of peak I_{sp} and finish burning at an O/F slightly above peak I_{sp}, the loss of total impulse can be kept to a few percent or less depending on the oxidizer. Generally, the maximum payload acceleration limit leads to a requirement that the oxidizer mass flow rate be throttled back while the port opens up, and the two effects tend to offset one another. For N_2O burning with paraffin, the exponent turns out to be very close to 0.5 and the O/F shift due to the opening up of the port is negligible.

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Throttling over a wide range is another matter. According to Eq. (2), the dependence of the O/F ratio on oxidizer mass flow rate is fairly strong. With oxygen, large O/F and hence, large I_{sp} variations do occur. This consideration can lead the designer to consider selecting a nitrogen-based oxidizer such as N_2O or N_2O_4 for which the sensitivity of I_{sp} to O/F near the peak I_{sp} is considerably less than that of LOx, although this would have to be balanced with the relative increase in oxidizer mass.

Although the hybrid enjoys many safety and environmental advantages over conventional rocket systems, large hybrids have never been commercially viable. The principal reason is the inherent low burning rate of conventional polymeric fuels due to the evaporative–diffusive nature of the combustion process. Traditional systems use polymeric fuels that evaporate too slowly, making it difficult to produce the high thrust needed for most applications.

To compensate, the surface area for burning must be increased, and as the scale of a hybrid rocket increases, the number of required ports also increases, leading to a fuel grain such as the wagon wheel design illustrated in Fig. 4. The need for multiple ports leads to a grain design with reduced volumetric loading and compromised structural integrity, a complex and ultimately impractical approach that has prevented the use of hybrids in launch applications where high thrust is required. The paraffin-based fuels discovered at Stanford University and developed by SPG promise to eliminate this disadvantage.

2. HISTORICAL PERSPECTIVE

The hybrid rocket concept has been around for more than seventy-five years. The first liquid propellant rocket launched by the Soviet Union was actually a hybrid that used liquid oxygen and gelled gasoline. The first flight of the 2.5 m long rocket occurred on Aug. 17, 1933. The motor produced about 500 N of thrust for 15 s, reaching an altitude of about 1500 m. As far as we know this was the first flight of a hybrid rocket. The rocket

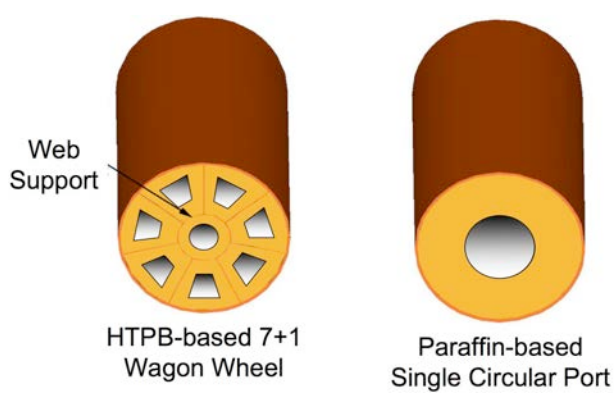

FIG. 4: Multiport vs single-port grain design.

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was designed by Tikhonravov, pictured in Fig. 5 with one of his designs, and built by a team from the Group for the Study of Reaction Motors (GIRD) that was headed by the legendary Korolev, who later would design the first Sputnik launchers.

One of the earliest efforts in the U.S. occurred at the General Electric main headquarters in Schenectady, NY, beginning in the late 1940s and continuing up to 1956. The company was interested in using hydrogen peroxide decomposed over a catalyst bed as a monopropellant. To augment the performance of the motor, they added a short extension containing polyethylene fuel to the combustion chamber between the catalyst bed and nozzle. The hot oxidizer reacted hypergolically with the fuel, increasing the specific impulse of the motor from 136 to 230 s (see Chap. 1 by Altman and Holzman in Chiaverini and Kuo, 2007).

Early hybrid rocket development began in earnest when flight test programs were initiated both in Europe and the U.S. in the 1960s. European programs in France and Sweden involved small sounding rockets, whereas the American flight programs were target drones [Sandpiper, High Altitude Supersonic Target (HAST), and Firebolt] that required supersonic flight in the upper atmosphere for up to 5 min. These latter applications were suitable for the conventional hybrid because its very low burning rate was ideal for a long duration sustainer operation.

Despite the very low regression rate of the fuel, in the late 1960s the Chemical Systems Division of United Technologies (CSD/UTC) investigated motor designs of larger diameters that could produce the high thrust required for space launch vehicles. They experimented with a 38 in. diameter motor delivering 40,000 lbs. of thrust. To achieve a

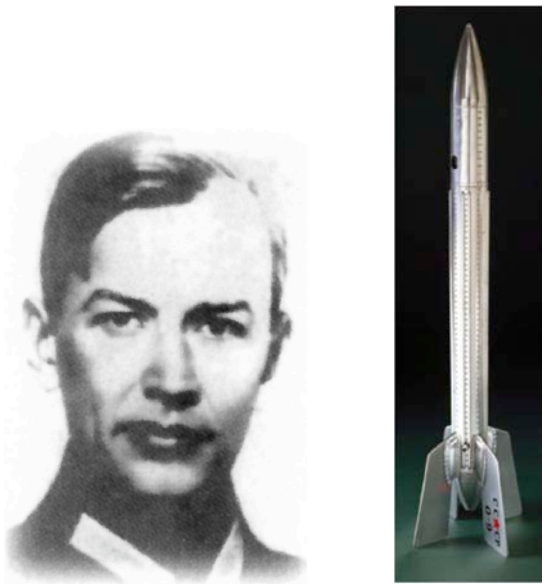

FIG. 5: Soviet rocket pioneer Tikhonravov and the GIRD-9.

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high mass flow rate, a motor with 12 ports in the fuel grain, each with its own injector, was required. Although the motor was successfully fired several times, it was recognized that the volumetric fuel loading efficiency was compromised, which would lead to a deficit in vehicle performance.

Interest in the hybrid was revived again in the late 1970s, when concern was expressed for the storage and handling of the large solid propellant segments of the Shuttle booster. The storage of potentially explosive grains is costly in terms of requirements for reinforced structures and interline distance separation between buildings. The same safety concern arose again after the January 1986 Challenger disaster, when it was recognized that a thrust termination option might have avoided the failure. This concern was heightened a few months later, when there was a Titan failure, caused by an explosion of one of the solid boosters.

Beginning in the late 1980s, two significant hybrid efforts occurred. One was the formation of the American Rocket Company (AMROC), an entrepreneurial industrial company devoted entirely to the development of large hybrid boosters. The second, with encouragement from NASA, was the formation of the Hybrid Propulsion Industry Action Group (HPIAG), composed of both system and propulsion companies devoted to exploring the possible use of hybrids for the Shuttle booster and other launch booster applications. Both efforts ran into technical stumbling blocks, caused by the low regression rate HTPB fuel that led to a large diameter motor with many ports needed to satisfy thrust requirements. The resulting configuration not only compromised potential retrofit for the Shuttle and Titan boosters but also raised questions about internal ballistic performance of a thin web multiport motor, especially toward the end of burning when grain structural failure can occur. The solution considered was to utilize a web support structure between ports; however, this only added to the concerns over volumetric loading efficiency. Although AMROC had many successful tests in 51 in. diameter motors, they ran into difficulties when the motor was scaled to 72 in. diameter and 250,000 pounds thrust. The low regression rate of the fuel dictated a 15-port grain design, leading to a low volumetric loading motor with grain integrity problems. In 1995, AMROC filed for bankruptcy.

Several hybrid propulsion programs were initiated in the late 1980s and early 1990s. The Joint Government/Industry Research and Development (JIRAD) program involved the testing of 11- and 24 in. diameter hybrid motors at the Marshall Space Flight Center. Another hybrid program initiated by AMROC during the early 90's was Defense Advanced Research Projects Agency's (DARPA's) Hybrid Technology Options Project (HyTOP). The goal of this program was to develop the HyFlyer launch vehicle and demonstrate the feasibility of hybrid boosters for space applications. The members of the HyTOP team were AMROC, Martin Marietta, and CSD/UTC.

The Hybrid Propulsion Demonstration Program (HPDP) began in March 1995. The goal of the HPDP was to enhance and demonstrate several critical technologies that are essential for the full-scale development of hybrid rocket boosters for space launch appli-

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cations. The government and industry participants in the program were NASA, DARPA, Lockheed Martin, CSD/UTC, Thiokol, Rocketdyne, Allied Signal, and Environmental Aeroscience Corporation. Even though the tasks of the HPDP program included systems studies and subscale testing, the main objective of the program was the design and fabrication of a 250,000 lb thrust test bed. The design of the motor was guided by the subscale motor tests performed under the JIRAD program. The multiport wagon wheel grain with seven spokes plus one center port (similar to Fig. 4) was made of conventional HTPB/Escorez fuel. The motor was fired for short times in July 1999. The motor exhibited large pressure oscillations and unequal burning rates in the various ports. Eventually, stable combustion in several long runs of the HPDP motor was achieved, but the underlying problem of low volumetric loading remained.

The most successful flight of a hybrid rocket occurred on Oct. 4, 2004 when Space Ship One reached an altitude of 100 km for the second time in a 1-week period to win the $10 million Ansari X-prize. The spacecraft used a four-port HTPB fueled motor and nitrous oxide oxidizer. Although this was a major success for hybrids, the performance required of the motor was far from that needed to lift a payload to orbit.

Throughout this history, the fundamental problem of low regression rate inherent in polymeric hybrid fuels remained unsolved. Moreover, it was clear that if a significantly higher burning rate could be realized for the hybrid motor, the difficulties mentioned above could be greatly reduced and a smaller, more efficient motor could be designed. This deficiency was recognized early on, and many attempts were made to increase the regression rate.

The most straightforward methods for enhancing the regression rate are based on increasing the heat transfer to the fuel. This can be accomplished in several ways: by adding energetic metals to increase radiation to the fuel surface, by adding particles to the fuel to roughen the fuel surface as it evaporates, or by segmenting the fuel with annular disks placed between segments designed to substantially increase turbulence levels in the port. Although increasing heat transfer to the fuel does increase the rate of evaporation, the increased mass flux from the fuel surface tends to reduce the temperature gradient at the surface, thus limiting the heat transfer rate. This is the well-known blocking effect in hybrids and limits the effectiveness of this approach to a maximum increase in regression rate of 30 to 50% compared to with the unenhanced case at the same mass flux (Karabeyoglu et al., 2001).

More than 100% increase in regression rate has been achieved by adding swirl to the flow along the port (Chiaverini and Kuo, 2007). It is difficult to maintain a high level of swirl over the full length of the port and it is unclear how to scale this approach to large motors. However, this approach can have application where extreme burning rates may be required.

DeLuca et al. (2009) showed that adding 11.2% by mass of alane more than doubled the regression rate of HTPB for oxidizer fluxes between 5.5 and 11.0 kg/m^{2}s. The oxidizer was a 50/50 mixture of oxygen and nitrogen. At elevated temperatures, alane

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rapidly dehydrogenates and video evidence suggests that hydrogen outgassing both increases fuel surface roughness and causes HTPB to be ejected from the fuel surface (Calabro et al., 2007). This is a promising approach, although alane adds cost and complicates fuel manufacture.

Probably the most widely used approach for increasing the regression rate of polymeric fuels is to add an oxidizing agent such as ammonium perchlorate (AP) to the fuel, thus turning it into a low-grade solid propellant. This has the effect of making the regression rate sensitive to both oxidizer mass flux and chamber pressure (as in a solid propellant rocket). This vastly enlarges the motor test space needed to accurately characterize the regression rate. The oxidizer mass flux in the port can vary by as much as a factor of 30 during the course of a burn and, unless a very accurate regression rate equation is available to the designer, there is no way to predict the motor performance. Moreover, the amount of AP needed for satisfactory effect approaches that required for a solid rocket. Adding oxidizing agents to the fuel removes the main safety advantage of the hybrid configuration.

In summary, the various attempts to increase the burning rate without compromising the safety and simplicity of hybrids have been largely unsuccessful until recently.

3. STANFORD RESEARCH

Research by the authors at Stanford University beginning in 1997 led to a class of very high regression rate fuels for use in hybrid rockets (Karabeyoglu et al., 2001, 2002; Karabeyoglu and Cantwell, 2002). This research indicated that hybrid rockets and/or combined cycle solid fuel ramjets (SFRJ) based on fuels of this type might have significant commercial applications. The new fuel produces a very thin (50–100 μm), low viscosity, low surface tension liquid layer on the fuel surface when it burns.

The instability of this layer illustrated in Fig. 6 is driven by the oxidizer gas flow in the port and leads to the entrainment of droplets into the gas stream greatly increasing the overall fuel mass transfer rate. This mechanism is modeled in Karabeyoglu et al. (2002), where the entrainment mass transfer rate is shown to depend inversely on the viscosity and surface tension of the liquid layer, as indicated in Fig. 6. In effect, this mechanism acts like a continuous spray injection system distributed along the port, with most of the fuel vaporization occurring around droplets convecting between the melt layer and the flame front. Because droplet entrainment is not limited by diffusive heat transfer to the fuel from the combustion zone, this mechanism is not limited by the blocking effect and leads to much higher surface regression rates than can be achieved with conventional polymeric fuels that rely solely on evaporation.

It was found that members of the normal-alkane class of hydrocarbons, which are solid at room temperature for carbon numbers greater than 14, have low surface tension and viscosity at the melt layer conditions typical of hybrid rockets (Karabeyoglu et al., 2005c). These fuels, which include the paraffin waxes and polyethylene waxes, are pre-

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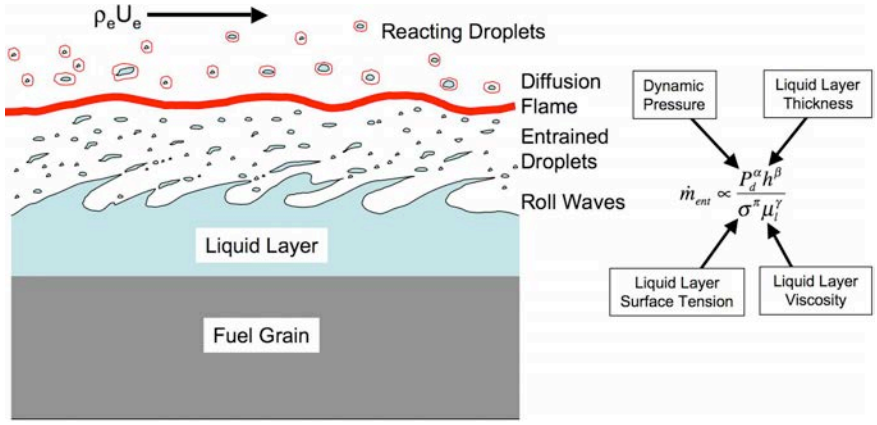

FIG. 6: Liquid layer instability and droplet entrainment mechanism; to the right is the dependence of entrained mass flow rate on free stream dynamic pressure and liquid layer properties.

dicted to have high regression rates at oxidizer mass fluxes covering a wide range of hybrid rocket applications. Making, handling, and transporting traditional solid rocket propellants is very costly, but the new paraffin-based fuel is nontoxic and nonhazardous and could actually be manufactured at the launch site (DeSain et al., 2009). No curing process is involved in fuel manufacture and the fuel has an essentially infinite shelf life. Paraffin is slightly viscoelastic. A study of the slump characteristics of the fuel showed that even for an Solid Rocket Booster (SRB)-sized motor the slump is negligible if the fuel temperature is maintained below 35°C (Kilic et al., 2003). The products of complete combustion of the new fuel with oxygen are simply carbon dioxide and water.

Regression rates 3–4 times the classical rate were first observed in a laboratory-scale motor in November 1998 using gaseous oxygen and a high melting point paraffin wax. The specific impulse of a paraffin-based hybrid motor is slightly higher than that of a kerosene-based liquid motor, and solid paraffin is approximately 17% more dense than kerosene. The n-alkanes comprise a wide range of molecular weight, surface tension, and viscosity and therefore can be used to create mixtures whose regression rate characteristics are tailored for a given mission.

The key fuel properties are low surface tension and low viscosity of the melt layer evaluated at the characteristic temperature of the layer. This forms the basis of a criterion that can be used to identify high regression rate fuels (Karabeyoglu et al., 2002). Not all fuels that form a melt layer at the fuel surface will entrain. For example, high-density polyethelene (HDPE), which is a conventional hybrid fuel, does form a melt layer, but the viscosity of the liquid is 4 orders of magnitude larger than paraffin, too viscous to permit droplet entrainment.

Recently, Nakagawa and his colleagues at Tokai University have visualized this mechanism in a combustion facility equipped with a quartz glass window (Nakagawa

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et al., 2009). The test section is 1 cm wide and 2 cm high. A frame from this video is shown in Fig. 7. The image clearly shows droplets that have been lifted off the fuel surface and are burning in the oxygen free stream. Some of the entrained droplets end up impinging on the glass side window and slowly convect along the window as they burn. The blue patch at the right end of the test section is the last vestige of one or more drops of fuel that earlier was burning on the window further upstream. The oxidizer mass flux in the experiments reported by Nakagawa et al. (2009) is very low and the test section operates at a low pressure of 1 atm. This makes it relatively easy to see the droplets. At the high mass fluxes and pressures that characterize hybrid rockets, the droplets would be much smaller and the combustion layer would be much thinner. In addition, the critical pressure of paraffin is only about 12 atm, well below typical rocket chamber pressures. The implication is that the surface of the droplets is likely to be very indistinct at typical chamber conditions. This is why the droplets sketched in Fig. 6 are depicted not as circular but as irregular in shape. The model that leads to the entrainment relation in Fig. 6 with the surface tension appearing in the denominator assumes subcritical conditions. In practice, the primary parameter characterizing the tendency for a fuel to entrain is the viscosity of the melt layer, which varies greatly from one fuel to another. The surface tension tends to be relatively constant for a wide range of fuels at subcritical conditions and is a less important discriminator.

4. SCALE-UP TESTS

The liquid layer entrainment hybrid rocket can be the basis for a system with a simpler grain design, reduced cost, reduced complexity, and increased performance, one that should be able to compete favorably with conventional solid or liquid systems. To further demonstrate the feasibility of the approach, a series of tests has been undertaken on intermediate-scale motors at pressures and mass fluxes representative of commercial

FIG. 7: Droplets entrained from the surface of paraffin burning with GOx from the video of Nakagawa et al. (2009).

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applications. A hybrid test facility designed to study these fuels was developed by NASA and Stanford researchers at NASA Ames Research Center and came online in September 2001. More than 40 tests were conducted in the period 2001–2003. The results of these tests compared with small motor tests at Stanford are shown in Fig. 8 (Karabeyoglu et al., 2004).

Recently, SPG has successfully conducted 40 LOx/paraffin-based hybrid motor firings using an 11 in. diameter, 7000 lb thrust motor. The primary focus of the testing effort has been to establish design guidelines for stable and efficient single circular port hybrid rocket systems using the cryogenic oxidizer LOx.

The SPG tests are illustrated in Figs. 9a and 9b with a representative image from two recent series of experiments. The measured C^* efficiency of the test depicted in Fig. 9a was 96%. In total, we have carried out well over 500 tests of paraffin-based fuels with several different oxidizers at several different scales. The main conclusions from the various scale-up tests are the following:

1. The regression rate behavior observed in the small-scale tests at Stanford prevails when the motor is scaled up to chamber pressures and mass fluxes characteristic of operational systems. Moreover, the regression rate data from large and small motors (Fig. 8) match quite well, indicating that the dependence of the regression rate on grain length is relatively weak and small-scale tests can be used to infer the behavior of larger motors. This result is extremely useful when it comes to developing the correct fuel formulation for a given mission.

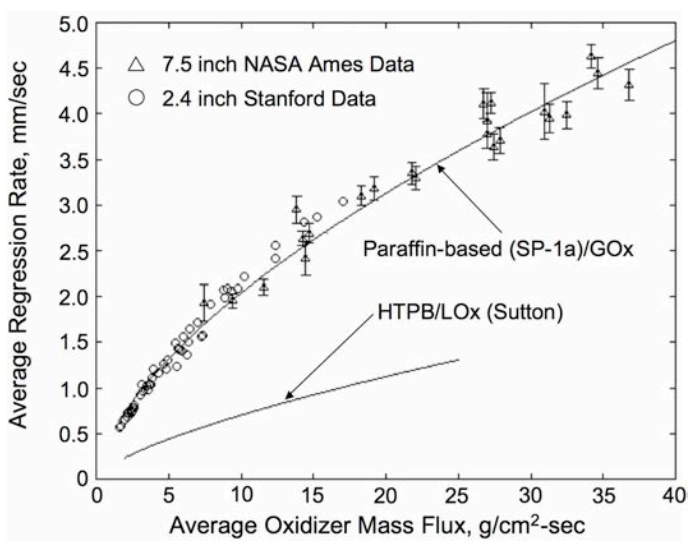

FIG. 8: Regression rate data from small-scale Stanford tests (left image; thrust class 50 lb, typical chamber pressure 200 psi) and medium-scale NASA Ames tests (right image; thrust class 3000 lb, chamber pressures up to 1050 psi) compared to HTPB.

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(a) (b)

FIG. 9: SPG tests: (a) Scale-up test using liquid oxygen. Thrust is approximately 7000 lb, grain diameter is 10 in. with a 5:1 converging–diverging nozzle. (b) Small-scale test using nitrous oxide. Thrust is approximately 150 lb, grain diameter is 2.7 in. with a 3:1 converging–diverging nozzle.

2. Paraffin-based fuels provide reliable ignition and stable combustion (for a properly designed motor) over the entire range of mass fluxes encountered (5–80 gm/cm^2 s) and with a variety of oxidizers.

3. For a carefully designed fuel/case system, paraffin-based fuels exhibit excellent structural integrity over the range of chamber pressures used (150–1050 psia).

5. ADDITION OF ENERGETIC METALS

Hybrids have a theoretical performance advantage over bipropellant liquid rockets in that the solid fuel makes it relatively easy to include energetic additives such as metal hydrides discussed earlier or high heat of reaction metals such as aluminum powder during casting. One effect of the aluminum is to increase the regression rate leading to a shorter grain design. Evans et al. (2003, 2005) used a direct x-ray technique to measure the effect of nanosized aluminum particles on enhancing the regression rate of HTPB and paraffin fuels. They observed up to a 60% increase in the regression rate for paraffin.

Figure 10 shows an analysis of the effect of aluminum addition on the vacuum specific impulse of paraffin reacting with dinitrogen tetroxide, N_2O_4. The figure shows two important effects. The first is that the maximum specific impulse increases slightly with increasing aluminum mass fraction. The second is that the peak in I_{sp} shifts to lower O/F ratio. The latter effect enables the designer to reduce the mass fraction of oxidizer. This simplifies the rocket hardware and reduces the size of the oxidizer tank. For higher aluminum mass fractions, this also means that there is an increase in the more dense propellant. Note that a 40% aluminum–60% paraffin mixture leads to a performance

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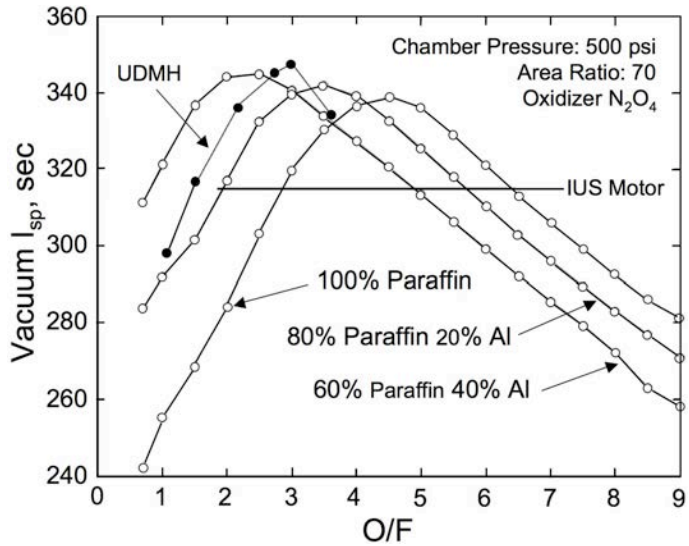

FIG. 10: Vacuum I_{sp} and O/F ratio for various concentrations of aluminum mixed with paraffin burning with N_2O_4.

comparable to unsymmetrical dimethylhydrazine (UDMH). Also indicated in the figure is the specific impulse of the Inertial Upper Stage (IUS). This is a two-stage solid propellant motor used successfully by Boeing from 1982 to 2004 on the Titan IV launch system. The figure suggests that a fully developed paraffin-based hybrid can be a competitive alternative to these solid and liquid systems. Finally, note that the O/F range near peak I_{sp} is considerably broader than in the case of oxygen (Fig. 3), suggesting a smaller I_{sp} change with throttling for N_2O_4.

There are some problems with aluminum addition. Fuel grain strength can be reduced and there can be losses associated with incomplete combustion of the aluminum and two-phase flow in the nozzle. Nanosized aluminum can be used to address some of these issues but increases complexity and cost.

6. GRAIN FABRICATION

The fuel composition needed for a given mission is produced at elevated temperature in liquid form. The liquid is then cooled and solidified to the required grain size and shape using a centrifugal casting process designed to produce crack-free and void-free grains despite the approximately 17% increase in fuel density that occurs during the solidification process. The low vapor pressure of the fuel lends itself to a safe, relatively simple fabrication process that can be carried out in a small-scale facility. No special ventilation is required. The fuel contains no toxic or oxidizing components and can be shipped by commercial freight as a nonhazardous commodity.

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7. A NEW CLASS OF OXIDIZERS BASED ON MIXTURES OF NITROUS OXIDE AND OXYGEN

Unlike the wide range of fuels that is currently available for chemical propulsion, the list of potential oxidizers is quite limited. Moreover, each oxidizer on this short list is associated with important shortcomings. For example, the high-density, storable oxidizer hydrogen peroxide, H_2O_2, can self-decompose explosively. The commonly used N_2O_4 gives good performance but is highly toxic. Inhibited red fuming nitric acid, IRFNA, is extremely toxic and somewhat corrosive.

Two of the most easily accessible liquid oxidizers that are widely used in rocket propulsion are liquid oxygen, O_2, and nitrous oxide, N_2O.

At room temperature (20°C), nitrous oxide is a low-density liquid with a specific gravity of 0.67 and a vapor pressure of about 850 psia and is commonly chosen for small rocket systems due to its self-pressurizing capability. Nitrous oxide is quite dense at temperatures in the range –80 to 0°C (specific gravity 1.25 to 0.9), although its vapor pressure is quite low at those temperatures. Nitrous oxide has a positive heat of formation and, when heated sufficiently or contaminated with hydrogen or hydrocarbon vapor, can decompose rapidly. Therefore, care is required in designing the delivery system for nitrous oxide, although it is many orders of magnitude more stable than hydrogen peroxide (Karabeyoglu et al., 2008).

Liquid oxygen is a commonly used oxidizer with a high specific impulse but has to be stored at deep cryogenic temperatures. The common features of both oxygen and nitrous oxide are their low toxicity and low cost compared to other oxidizers. While each has shortcomings, mixing the two helps to alleviate the shortcomings of both.

SPG engineers have recently developed a new class of oxidizers based on refrigerated mixtures of nitrous oxide and oxygen called Nytrox (Karabeyoglu, 2007, 2009). The basic idea is to combine the high vapor pressure of dissolved oxygen (20 to 120 atm) with the high density of refrigerated nitrous oxide to produce a safe, nontoxic, self-pressurizing oxidizer with high density and good I_{sp} performance.

The main idea is summarized in Fig. 11. Figure 11a shows how the vapor pressure of a Nytrox mixture depends on the mole fraction of oxygen in the liquid and vapor phase at a given temperature. As increasing amounts of oxygen are dissolved in the N_2O liquid, the vapor pressure varies from 20 to 120 atm. This is in the useful range for a rocket propulsion system. Note that at mixture equilibrium the vapor phase is mostly oxygen, greatly reducing the possibility of N_2O decomposition in the oxidizer tank ullage (Karabeyoglu et al., 2008).

Figure 11b shows the relation between liquid density and vapor pressure for several oxidizers including LOx, N_2O, and mixtures of N_2O and oxygen at various temperatures. Notice that the densities and vapor pressures of the pure substances are very sensitive to temperature. This figure illustrates why warming LOx is not an effective means of pressurization; the density decreases too rapidly. Adding oxygen to refrigerated N_2O in-

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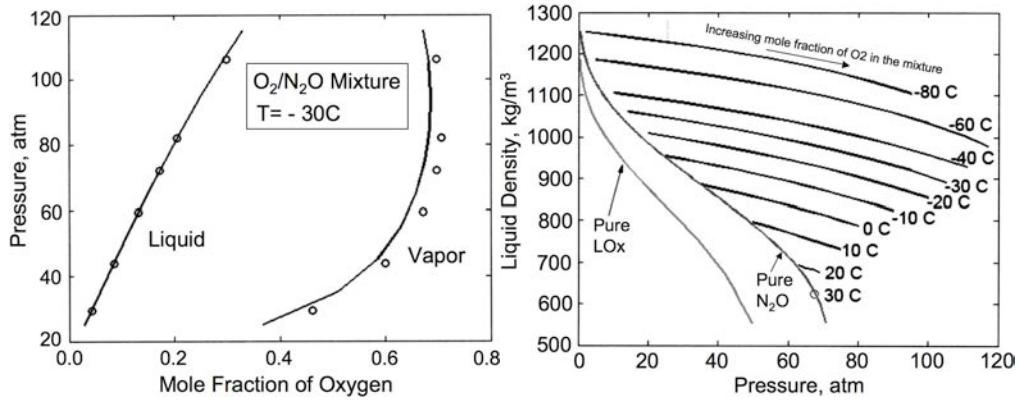

(a) (b)

FIG. 11: (a) Nytrox vapor pressure vs oxygen mole fraction at –30°C. (b) Liquid density vs vapor pressure for Nytrox mixtures from Karabeyoglu (2009).

creases the vapor pressure of the mixture with only a relatively small decrease in liquid density at fixed temperature, as indicated in Fig. 11b. The –30°C line in Fig. 11b corresponds to the liquid line in Fig. 11a. At that temperature, the maximum mole fraction of oxygen that can be dissolved in the liquid is about 27%.

Figure 12 shows the performance of paraffin-based fuels with several oxidizers: LOx, N_2O_4, N_2O, and Nytrox with various mass percentages of oxygen. The percent of dissolved oxygen and the vapor pressure of the mixture, 500 psi in the case shown, is determined by the selected refrigeration temperature.

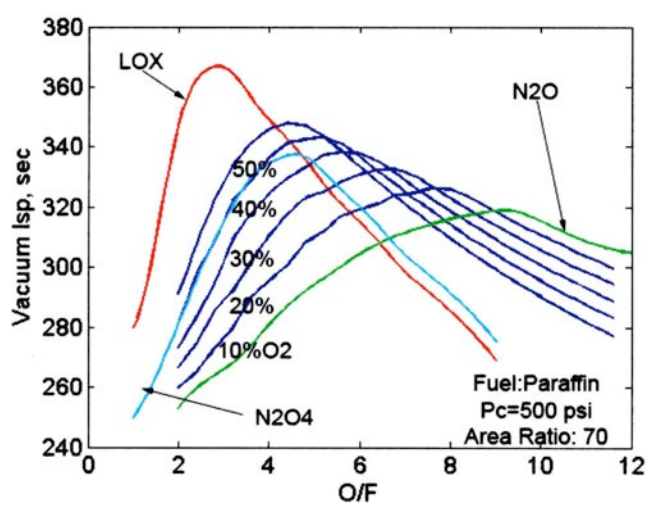

FIG. 12: I_{sp} performance of Nytrox with Paraffin fuel compared to pure oxidizers from Karabeyoglu (2009).

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At low temperatures, the performance is excellent. For example, a 35% O_2/N_2O mixture at –80°C and 100 atm matches the peak theoretical I_{sp} performance of N_2O_4. The self-pressurizing feature substantially reduces the size of the required pressurization system, leading to reduced overall system size and weight.

To pursue this issue further, a detailed system performance study was carried out comparing two hybrid designs with a widely used commercial launch strap-on booster, the GEM-40 manufactured by ATK. The structural mass margins used were 10% and the general design methodology was similar to the approach used in Karabeyoglu et al. (2005c). The hybrids are constrained to deliver the same total impulse over the same burn time with the same vehicle diameter as the GEM-40. The results are presented in Table 1. The high oxidizer density, ability to use composite tanks (because of the noncryogenic operation), and relatively high optimal O/F ratio of the Nytrox system lead to a lighter, more compact vehicle compared to a LOx/paraffin system. Whereas the Nytrox design is longer than the GEM40, it is slightly lighter. In terms of delivered impulse, the higher density of the solid motor is more than compensated by the higher specific impulse of the Nytrox system.

TABLE 1: Comparison of an aluminized paraffin-Nytrox (–80°C, 20% aluminum) hybrid with a commonly used solid booster (GEM40) and a LOx/paraffin hybrid from Karabeyoglu (2009)

System	**GEM40**	**Nytrox80/20 Al-paraffin**	**LOX/paraffin**
Matched parameters			
Total impulse, N s	3,162,107	3,162,107	3,162,107
Outside diameter, m (in)	1.02 (40.0)	1.02 (40.0)	1.02 (40.0)
Burn time, s	63.3	63.3	63.3
Systems study summary—length and weight			
Overall length, m (in)	**13.00 (511.8)**	**18.24 (718.2)**	**19.69 (775.0)**
Propulsion system mass, kg	**13,080**	**12,987**	**13,233**

8. CONCLUSION

The inherent advantages of the hybrid rocket concept include increased safety, reduced environmental impact, throttling, shutdown/restart, addition of metals or energetic materials, and lower life cycle costs. The study discussed in Section 7 lends support to our contention that a high regression rate paraffin-based hybrid can achieve a compact, single-port design with performance that competes favorably with solid and liquid systems. Compared to conventional oxidizers, Nytrox provides further safety and cost improvements without sacrificing good performance. Together these two technologies can lead to game-changing propulsion solutions for a wide variety of launch applications.

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There is probably no other good idea in chemical rocket propulsion that has had as long a development as the idea of the hybrid rocket. In their 1979 historical review, Mukunda et al. (1979) suggested that the failure to develop the hybrid rocket was largely a historical consequence of the fact that solids and liquids were developed to a high technical level first, leaving little need to develop a third option. However, there were also significant technical hurdles that needed to be overcome to produce a design that could provide competitive performance. In the years following the Challenger disaster there was a clear recognition of the advantages of hybrid technology by government and industry, especially NASA, but by the end of the 1990s, interest in hybrids was clearly on the wane as the HPDP program was beginning to wind down with many technical challenges still unsolved.

The last decade has seen important technical advances and renewed interest in hybrid rockets. In the U.S., there is active research going on at SPG, Stanford, Penn State, Purdue, Sierra Nevada, the Aerospace Corporation, and NASA. At NASA Ames, a large hybrid sounding rocket called Peregrine (Doran et al., 2009) is under development. Presently, SPG is testing 11 in. diameter motors (Fig. 9) and preparing to test a 24 in. diameter LOx/paraffin motor in the 35,000 lb thrust class. The largest flight motor under development is the propulsion system for Space Ship Two currently being tested by a Sierra Nevada Corporation/Scaled Composites team and Virgin Galactic. In the near future this system will power the emerging space tourism industry.

At the yearly U.S. Joint Propulsion Conference, one can count a large number of programs outside the U.S. including in countries such as Japan, Italy, France, Germany, Israel, South Korea, Brazil, New Zealand, and Norway. Today there is every reason to expect that hybrid rocket technology will finally be developed to an advanced level. With the new technology will come new opportunities to design environmentally clean, low-cost, high-performance, hybrid propulsion systems for a wide variety of launch applications.

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