

# Nontoxic Orbital Maneuvering and Reaction Control Systems for Reusable Spacecraft

Eric Hurlbert,\* John Applewhite,† and Tien Nguyen‡

*NASA Johnson Space Center, Houston, Texas 77058*

Brian Reed‡

*NASA Lewis Research Center, Cleveland, Ohio 44135*

and

Zhang Baojiong§ and Wang Yue§

*Shanghai Academy of Spaceflight Technology, Shanghai 200233, People's Republic of China*

Toxic propellants have a high ground operations cost because of the potential hazards that require extensive safety precautions, particularly for reusable spacecraft. Nontoxic propellants for orbital maneuvering and reaction control systems have received periodic attention since the late 1960s as new reusable vehicles and upgrades to existing vehicles are proposed. This paper discusses the spacecraft requirements that drive propellant selections, the viable candidates for nontoxic propellants, and the system concepts and technologies required. Options for nontoxic propellants are also discussed, which are categorized as monopropellants, storable bipropellants, and cryogenic oxygen-based bipropellants. Monopropellants provide inherently simple systems and are most suited to low total impulse systems. Hydrogen peroxide and kerosene is a promising storable bipropellant in terms of its density, specific impulse, and low toxicity for long-duration spacecraft on-orbit propulsion systems. This combination can be made hypergolic, which renders it a very effective replacement for currently used storable propellants. The hypergolic characteristics of kerosene with additives and high-concentration hydrogen peroxide are presented in detail. Higher-performance liquid oxygen and alcohol or hydrocarbon fuels are advantageous for reusable propulsion systems that emphasize fluid commonality with other spacecraft systems and for human exploration missions where in-situ propellant production is foreseen. The prospects for further research work on all of these propellants are also discussed.

## I. Introduction

**H**YDRAZINE, nitrogen tetroxide, and monomethyl hydrazine have satisfied the requirements of many spacecraft since the 1960s; however, the use of these toxic and corrosive propellants in reusable spacecraft has not been entirely successful. While flight performance has been adequate, these propellants present many ground safety hazards because of their toxic and corrosive properties. As environmental protection and safety requirements are increased, the use of nontoxic propellants in expendable spacecraft, including satellites, may become inevitable. This paper discusses the spacecraft requirements that drive propellant selections, the viable candidates for the nontoxic propellants, and the system concepts and technologies required.

Monopropellants provide inherently simple systems and are most suited to low total impulse systems. Hydroxyl ammonium nitrate- (HAN-) based propellants, gaseous nitrogen mixed with low concentrations of oxygen and hydrogen, and hydrogen peroxide ( $H_2O_2$ ) all provide viable solutions. The current status of these propellants is discussed. Many of these appli-

cations are also candidates for electric propulsion or solar thermal propulsion.

For storable bipropellants, hydrogen peroxide and kerosene are promising in terms of their high density specific impulse, long-term storage capability, and low toxicity for spacecraft on-orbit propulsion systems. If this combination can be made hypergolic, then they become a very effective replacement for storable propellants. Through extensive laboratory testing a promising kerosene additive capable of causing the self-ignition of kerosene and  $H_2O_2$  has been identified. Test-firings have been conducted on the small test-piece thrust chamber. Encouraging progress in research on the key technology of the self-ignition has been achieved. The hypergolic characteristics of kerosene with additives and high-concentration  $H_2O_2$  are presented in detail. The prospects for further research on all of these propellants are discussed.

Liquid oxygen (LOX) with alcohol or a hydrocarbon fuel is currently being pursued for the Space Shuttle Orbiter. This combination offers higher performance, spacecraft fluid system commonality, and very low toxicity. This is a key element of a propulsion technology road map that uses oxygen-based propulsion for the Orbiter while incorporating the future needs of other human exploration and development of space (HEDS) activities. In-situ resource utilization (ISRU), which manufactures oxygen from resources on the moon and Mars, is viewed by some as an enabling technology for HEDS.<sup>1</sup> Concepts for implementing an oxygen and ethanol reaction control system on the Orbiter are presented.

## II. Spacecraft Requirements and Propellants

A variety of spacecraft requirements, graphed in terms of mission life, total impulse, and thrust level, is shown in Fig. 1. These are the main drivers of propellant selection for orbital

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\*Aerospace Technologist, Energy Systems Division. E-mail: eric.a.hurlbert@jsc.nasa.gov.

†Aerospace Technologist, Energy Systems Division.

‡Aerospace Technologist, Onboard Propulsion Branch.

§Professor, 801 Research Institute.

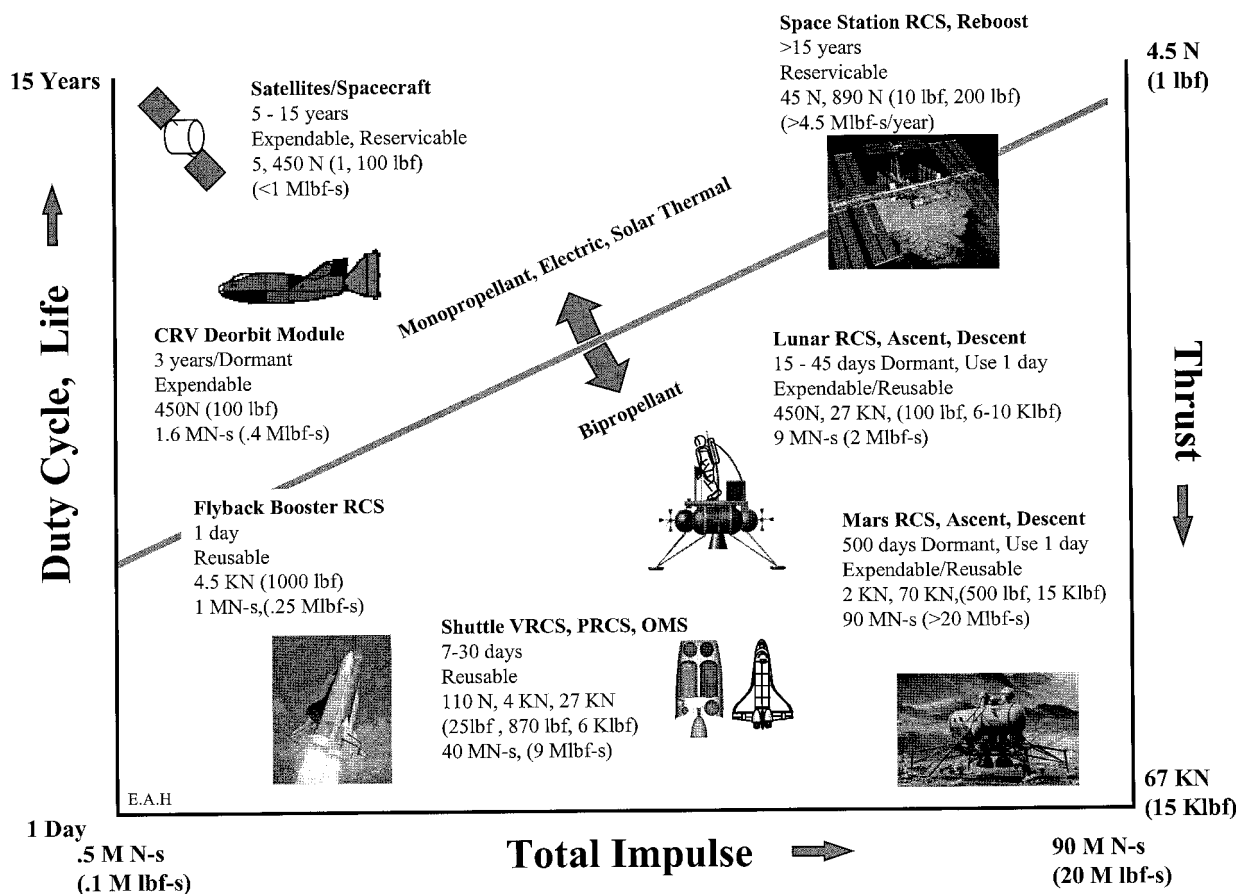


Fig. 1 Total impulse, mission life, and thrust level requirements for various spacecraft.

maneuvering and reaction control systems (RCS). Table 1 suggests propellants for given applications. The philosophy is to use the simplest propellant combination that will provide the necessary performance. If a cold-gas system will work, then it is recommended. Cold-gas or warm-gas systems are suitable for very low total impulse, low thrust, and short-to-long life systems, such as tele-robotic freeflyers and crew maneuvering systems. Satellites occupy the region of long life of 5–15 years, low thrust, and low total impulse. Satellite systems are usually expendable or serviceable. Electric and solar thermal propulsion are also making significant progress for these applications.

The deorbit module for the X-38 space station crew return vehicle (CRV) is currently designed to be separate and expendable to keep the toxic propellants out of the spacecraft internal subsystems. The deorbit module is dormant for up to 3 years and provides rapid power-up response. Hydrazine is currently the low-cost state-of-the-art choice. Monopropellants, such as HAN or  $H_2/O_2$ , would be the choices for future upgrades or operational vehicles. The X-38 does use cold gas for the spacecraft RCS.

Space stations require large quantities of propellants to counteract the drag forces in low Earth orbit. Nontoxic propellants would be much safer in the vicinity of human operations. Hydrogen peroxide and ethanol is a good storable combination for space stations. Ethanol provides clean combustion and will not contaminate external surfaces. LOX is also a potential oxidizer. The LOX oxygen boil-off could be used for crew breathing or it could be reduced, or possibly eliminated, using cryocoolers.

Liquid flyback boosters (LFBB), lunar landers, and space shuttles are short-duration, high-thrust applications. These applications are a primary focus of this paper. The Space Shuttle Orbiter requires reusability, high total impulse, high thrust, and short mission life (7–30 days). LOX with a hydrocarbon or

alcohol fuel best satisfies these requirements. A flyback booster requires reusability, a short mission life (1 day), high thrust, and low total impulse. Reusable launch vehicles (RLV), such as single-stage-to-orbit and Lunar/Mars transtage main engines, may require the higher performance of oxygen and methane or hydrogen. One goal of a nontoxic propellant system for the Space Shuttle is to provide system and/or component commonality across these applications and others, such as landers.

#### A. Monopropellants

Nontoxic monopropellant propulsion systems are being developed for small spacecraft (less than 50 kg), as a replacement for cold nitrogen gas ( $GN_2$ ) and monopropellant hydrazine ( $N_2H_4$ ) propulsion. Small spacecraft have limited propulsion options because volume and power are usually limited.  $GN_2$  propulsion offers the simplest and lowest-cost option, with essentially no power requirements or special ground-handling procedures. However,  $GN_2$  has low specific impulse ( $I_{sp}$ ) performance (60 s) and storage density ( $\sim 0.23$  g/cc). Warm gases are nondetonable mixtures of stoichiometric hydrogen and oxygen diluted with an inert gas, such as nitrogen or helium. The mixture can be safely stored as if it were a pressurized inert gas. Passing the mixture through a catalyst bed, however, will cause the hydrogen and oxygen to react, and the mixture will be heated. This provides a propulsion system of simplicity comparable to cold-gas systems, but with nearly double the  $I_{sp}$ .

Warm gas mixtures are often referred to as Tridyne<sup>TM</sup>. Rocketdyne conducted much of the initial technology work for these mixtures in the 1960s. Tridyne mixtures were investigated for both pressurization<sup>2</sup> and propulsion<sup>3</sup> applications. More recently, a warm-gas system was developed for pressurization of the LOX tank on a single-stage hybrid rocket.<sup>4</sup> There is currently a research effort to develop a miniature (0.11-N) warm-gas thruster for microsatellite applications.<sup>5</sup> A range of catalyst

Table 1 Suggested propellants for various space vehicle application

Application	Increasing performance capability									
	GN <sub>2</sub>	Tridyne	N <sub>2</sub> H <sub>4</sub>	HAN based	NTO, MMH	H <sub>2</sub> O <sub>2</sub> ethanol	O <sub>2</sub> ethanol	O <sub>2</sub> RP1	O <sub>2</sub> methane	O <sub>2</sub> H <sub>2</sub>
Crew maneuver	□									
CRV RCS	□	□		□						
Transtage RCS			□	□	□	□				
Lander RCS			□	□	□	□	□	□	□	
Flyback booster RCS		□	□	□	□	□	□	□	□	
CRV deorbit			□	□	□	□				
Space station			□	□	□	□	□			□
Orbiter RCS			□	□	□	□	□	□		
Orbiter OMS					□	□	□	□		
RLV RCS							□	□	□	□
Lander Asc/Des							□	□	□	□
RLV OMS									□	□
Transtage main										□

materials were evaluated in this program, using heavyweight thruster hardware and an 88% nitrogen/8% hydrogen/4% oxygen mixture. A flight-weight thruster was tested with this mixture, demonstrating an  $I_{sp}$  of 130 s. Higher-performance warm-gas mixtures can be formulated by increasing the oxygen content while maintaining the stoichiometric mixture ratio.

Monopropellant N<sub>2</sub>H<sub>4</sub> offers significantly higher performance ( $I_{sp} = 223$  s) and storage density (1.0 g/cc) with modest power requirements (for catalyst bed heaters). However, N<sub>2</sub>H<sub>4</sub> is toxic, carcinogenic, and flammable, requiring extensive infrastructure and ground-handling procedures. For small spacecraft, the costs associated with using N<sub>2</sub>H<sub>4</sub> are a significant percentage of the overall propulsion system costs. A class of monopropellants based on HAN offers the performance and density of N<sub>2</sub>H<sub>4</sub>, with the safety and handling benefits of GN<sub>2</sub> systems. HAN-based formulations were pursued for many years by the U.S. Army as liquid gun propellants (LGP).<sup>6</sup> The U.S. Army LGP program developed these propellants as environmentally benign, insensitive munitions.

The propellant formulations consist of an oxidizer-rich salt (HAN) and a fuel-rich salt, dissolved in water. The U.S. Army development focused on the two HAN-based formulations, LP1846, which uses triethanol ammonium nitrate (TEAN) as the fuel salt, and LP1898, which uses diethyl hydroxyl ammonium nitrate (DEHAN). Both formulations consist of approximately 61% HAN, 20% water, and 19% fuel. Because HAN and water make up the majority of the formulations, safety and compatibility data for LP1846 and LP1898 will probably be applicable to all HAN-based monopropellants. HAN-based propellants, primarily LP1846, have been the subject of numerous health and safety studies and, to date, all data collected have been favorable. The generant and exhaust are benign. Water-repellent materials and elastomeric gloves are sufficient for handling. LP1846 has proven negative as both a carcinogen and mutagen. No inhalation hazards are associated with these propellants (unless aerosolized) or with their exhaust products (carbon dioxide, nitrogen, and water). The propellant has proven to be inflammable and insensitive at atmospheric pressure.

Although HAN-based monopropellants have a wide liquid-temperature range, thermal management will be driven by the viscosity. LP1846 and LP1898 experience dramatic variations in viscosity at temperatures below  $-30^{\circ}\text{C}$ .<sup>6</sup> This represents the lower practical temperature limit for HAN-based monopropellants. Note that N<sub>2</sub>H<sub>4</sub> freezes at  $0^{\circ}\text{C}$  and is generally maintained at a minimum temperature of  $10^{\circ}\text{C}$ . The thermal management of HAN-based monopropellant systems, then, is likely to be simpler than for N<sub>2</sub>H<sub>4</sub> systems. The performance of a HAN-based monopropellant formulation depends on its water content and selection of the fuel component. Decreasing the amount of water increases the exhaust temperature and the  $I_{sp}$ . However, low water content formulations may be less sta-

ble and more impact sensitive. Furthermore, decreasing the water content increases the molecular weight of the exhaust, which detracts somewhat from the performance gains caused by increased temperature.

The fuel ingredient trade involves maximizing the heat of formation while minimizing the molecular weight of the exhaust. Conservatively assuming an overall efficiency of 85%,  $I_{sp}$  of 215 and 223 s can be achieved (at 50:1 area ratio and 690 kPa chamber pressure) for LP1846 and LP1898, respectively.<sup>7</sup> Decreasing the water content in these formulations by 5% increases  $I_{sp}$  by about 10 s. The densities of the U.S. Army LGP formulations are similar: 1.43 and 1.39 g/cc for LP1846 and LP1898, respectively. Because HAN and water will make up the majority of any formulations, the density of a HAN-based monopropellant can be estimated as 1.4 g/cc. This is a 40% increase in density over N<sub>2</sub>H<sub>4</sub>, and so a HAN-based monopropellant with similar  $I_{sp}$  performance to N<sub>2</sub>H<sub>4</sub> will have an advantage in volume-limited applications.<sup>8</sup>

The major technology challenge for HAN-based monopropellant rockets is ignition. Catalytic ignition (as used with N<sub>2</sub>H<sub>4</sub>) is the preferred method, although HAN-based monopropellants produce a more severe combustion environment than N<sub>2</sub>H<sub>4</sub>. The HAN-based monopropellants produce higher molecular-weight exhaust gases ( $\sim 24$  g/mol) than N<sub>2</sub>H<sub>4</sub> exhaust (14 g/mol). Therefore, HAN-based monopropellants have to produce higher exhaust temperatures than N<sub>2</sub>H<sub>4</sub> (1650 vs  $980^{\circ}\text{C}$ ) to maintain the same  $I_{sp}$  level. Shell 405 (iridium on alumina) is the established catalyst material for N<sub>2</sub>H<sub>4</sub>. Above  $1000^{\circ}\text{C}$  the noble metal in these catalysts will begin to sinter, resulting in the loss of chemically active surface area. Furthermore, alumina will undergo a phase transformation above  $1000^{\circ}\text{C}$ , resulting in cracking and loss of structural support. While there are a limited number of catalyst materials for operation above  $1650^{\circ}\text{C}$ , a broader range of catalyst options are available if propellant formulations are reduced below  $1400^{\circ}\text{C}$ . The lower-temperature formulations, however, will have lower  $I_{sp}$  performance than N<sub>2</sub>H<sub>4</sub>. There are at least three development paths, then, for HAN-based monopropellant engines.

1) High-performance formulations ( $I_{sp} > 220$  s) with established catalyst materials, but restricted to limited duty cycle applications such as orbit insertion of satellites that usually require less than six firings and three h total duration.

2) Lower-performance formulations ( $I_{sp} = 190\text{--}220$  s) focusing on near-term development of catalyst materials for the demanding duty cycles of on-orbit propulsion ( $>100,000$  cycles, up to 10-h total duration).

3) High-performance formulations with high-temperature catalyst materials or alternative ignition methods that will probably entail a longer-term development, but would be applicable across the range of propulsion functions.

Current NASA-supported development efforts have concentrated on the second path, that is, near-term development of a

0.22-N monopropellant engine.<sup>9</sup> An extensive matrix of propellant formulations and catalyst materials is being evaluated in laboratory-scale testing. The high density of HAN formulations make them attractive for volume-limited spacecraft, even at the reduced  $I_{sp}$ . Also, ground-handling issues are more important for small spacecraft, where the only other non-toxic option is cold-gas propulsion. A high-performance, low-duty cycle development is not currently supported by NASA, although a critical demonstration was performed. Using pyrotechnic ignition and heavyweight hardware, a HAN-based monopropellant was tested at chamber pressures from 2.5 to 4.8 MPa. A combustion efficiency of 95% was achieved in testing, projecting to an  $I_{sp}$  of 270 s at an area ratio of 50:1. High-temperature catalyst materials and alternative ignition methods are being investigated as part of a longer-term technology program. When ignition methods for high-temperature formulations are demonstrated, the development of long-life, high-performance HAN-based monopropellants can proceed.

## B. Storable Bipropellants

The characteristics of various bipropellants are summarized in Table 2 as they relate to criteria important to orbital maneuvering system (OMS) and RCS. These are all relative comparisons of the propellant combinations for each given criterion. A plus symbol indicates that the system is desirable or advantageous for that criterion relative to the other systems, and a negative symbol indicates a disadvantage of the system compared with the others, with respect to that criterion. The final selection of a propellant depends on the relative importance of each criterion to a particular application.

Hydrogen peroxide ( $H_2O_2$ ) and ethanol or kerosene make an attractive Earth-storable bipropellant combination. Concentrations in the range of 90–98% are most applicable to liquid-

propulsion systems. The high density  $1430 \text{ kg/m}^3$  ( $90 \text{ lbm/ft}^3$ ) and storability of  $H_2O_2$  make it an option for long-duration missions. Materials selection and system cleanliness are keys to storage life. The performance ranges from 295 to 305 s for an OMS engine. The U.S. Air Force and Navy have considered  $H_2O_2$ /kerosene for bipropellant rocket-propulsion systems.<sup>10,11</sup> Applications such as a spaceplane are discussed by Wilkinson and Bentz,<sup>12</sup> and Clapp and Hunter.<sup>13</sup> NASA has also considered the possible use of  $H_2O_2$  and hydrocarbons on advanced spacecrafts. In the HL-20 Manned Orbiter program,<sup>14</sup>  $H_2O_2$ /kerosene was selected as the propellant combination for the OMS and RCS based on a comparison among nitrogen tetroxide (NTO)/monomethyl hydrazine (MMH), NTO/ $N_2H_4$ ,  $LO_2/CH_4$  and  $LO_2/LH_2$ .

The theoretical performance characteristics of  $H_2O_2$ /kerosene with respective concentrations of  $H_2O_2$  of 95 and 98% are shown in Table 3. Table 3 also presents the performance characteristics of several Earth-storable and cryogenic propellant combinations used for propulsion-system evaluation. Performance is calculated by assuming one-dimensional equilibrium flow at a chamber pressure of 1 MPa (150 psia). The calculation takes into account the combustion efficiency, boundary-layer losses, heat transfer losses, divergence losses, and chemical kinetics losses in the nozzle flow. The actual performance of the OMS engine is 93% of the theoretical performance for one-dimensional equilibrium flow with a nozzle area expansion ratio of 50. The performance of the RCS thruster is 86% of the theoretical performance for one-dimensional equilibrium flow with a nozzle area ratio of 25 because of the necessity of taking into account additional degradation in performance caused by pulse mode startup and shutdown and for fuel film cooling.

In Table 3, the comparative specific impulse and density specific impulse performances are given against those of NTO/

Table 2 Evaluation of bipropellants for a reusable OMS/RCS

	MMH/NTO pressure-fed	$H_2O_2$ /H-C pressure-fed	LOX/ alcohol pressure-fed	LOX/H-C pressure-fed	LOX/ methane pump-fed	LOX/ $LH_2$ pump-fed
Performance						
Propellant mass, $I_{sp}$	—	—	—	—	+	+
Power required	+	+	+	+	+	+
Volume, density $I_{sp}$	+	+	+	+	+	—
Cost						
Inert (dry) mass	+	+	+	+	—	—
Number of components	+	+	+	+	—	—
Integration with power	—	—	+	+	+	+
Integration with ECLSS	—	—	+	+	+	+
Thermal management	+	+	—	—	+	—
Residues and soot	—	+	+	—	+	+
Materials (corrosivity, temperature)	—	—	+	+	+	—
Leakage	—	+	+	+	+	—
Number of required instruments	+	+	+	+	—	—
Number of flight operations, flight rules	+	+	+	+	—	—
Toxicity	—	±	+	+	+	+
Propellant cost	—	—	+	+	+	—
Flammability	—	+	+	+	+	—

Table 3 Bipropellant properties

	Mixture ratio	OMS $I_{sp}$ s	RCS $I_{sp}$ s	Oxidizer density, $\text{kg/m}^3$ ( $\text{lbm/ft}^3$ )	Fuel density, $\text{kg/m}^3$ ( $\text{lbm/ft}^3$ )	OMS density $I_{sp}$ $\text{KN-s/m}^3$ (K $\text{lb-f-s/ft}^3$ )
90% $H_2O_2$ + no. 3 aviation kerosene	7.70	295	264	1388 (87)	800 (50)	377 (24.4)
96% $H_2O_2$ + no. 3 aviation kerosene	7.10	305	272	1430 (89)	800 (50)	397 (25.7)
NTO/MMH	1.65	310	279	1447 (90)	872 (55)	362 (23.8)
NTO/MMH	2.40	317	282	1447 (90)	872 (55)	386 (24.1)
LOX/ethanol	1.8	323	283	1140 (71)	788 (49)	328 (20.5)
LOX/kerosene	2.6	334	292	1140 (71)	800 (50)	349 (21.8)
LOX/methane	3.3	344	305	1140 (71)	415 (26)	—

MMH. The bulk density of 90%  $\text{H}_2\text{O}_2/\text{HC}$  is higher than that of NTO/MMH because of the high mixture ratio. However, the high mixture also causes the oxidizer tank to be much larger than the fuel tank. The density specific impulse of the propellant combination is higher, even though its specific impulse is lower than that of NTO/MMH by 10–20 s. This is attractive when volume is an important criterion for propellant selection. In brief, the mass of the 90%  $\text{H}_2\text{O}_2$ /kerosene system is 5–10% heavier than that of NTO/MMH, but the volume is 5–10% smaller. If the 96% concentration  $\text{H}_2\text{O}_2$  is adopted the entire system weight will be comparable to that of the NTO/MMH.

In a comparison of candidate propellant combinations for on-orbit propulsion systems, factors in addition to mass and volume, such as toxicity, may become important and must be taken into account. Toxicity affects the cost of operation and maintenance. The main advantages of nontoxic propellants are their environmental friendliness and harmlessness to personnel. Although  $\text{H}_2\text{O}_2$  has a threshold limit value (TLV) of 1 ppm, it is only an irritant and has a very low vapor pressure 0.27 Pa ( $\sim 0.04$  psia). A ranking among candidate propellant combinations was made using an analytical hierarchy process with relative weights of 0.2 for mass, 0.2 for volume, and 0.6 for toxicity.<sup>15</sup> The highest score was assigned to  $\text{H}_2\text{O}_2$ /kerosene. It should be pointed out that the choice of the relative weights is somewhat arbitrary and should be based on the needs of the particular application.

The hypergolic characteristics of NTO/MMH make the system very simple and reliable. If  $\text{H}_2\text{O}_2$  and hydrocarbon fuels are to replace NTO/MMH easily, the propellant combination must be rendered hypergolic. This can be done in two ways. One method is to trigger the combustion reaction using a torch ignitor with decomposed  $\text{H}_2\text{O}_2$  and fuel. Once the reaction starts, the ignitor can be shut off because of the self-sustainability of  $\text{H}_2\text{O}_2$ /hydrocarbon reactions. This method is feasible for OMS, but may not be suitable for the RCS that requires short impulse operations. In this case a dual-mode propulsion system can be used: a monopropellant catalyst decomposition system for RCS with a bipropellant system for OMS. Of course, the lower specific impulse of a monopropellant leads to a larger quantity of propellant loading for a fixed total impulse requirement. The effective life of the catalyst bed is another concern. The second method is to mix an additive into the hydrocarbon to make it self-ignitable with  $\text{H}_2\text{O}_2$ . This makes it possible to adopt a unified system with improved mass and volume characteristics. The problem with catalyst-bed life can also be avoided. Thus, the development of a self-ignition system for  $\text{H}_2\text{O}_2$ /kerosene becomes enabling for its utilization for spacecraft on-orbit propulsion system.

Early in the World War II period, the Germans developed fuels capable of self-ignition with  $\text{H}_2\text{O}_2$ . The fuels used at that time were alcohols, such as Ergol 56 (62.5% methyl alcohol, 30% catechol, and 7.5% ferrous dialkyl naphthyl sulfonate) and C-Stoff (70% methyl alcohol and 30% hydrated hydrazine).<sup>16</sup> In addition to methyl alcohol, the catalyst and promoter accounted for about 40% of the mixture, which considerably affected the operation and energy characteristics of the propellant. In the U.S. in the late 1940s, exploration of self-igniting fuels with  $\text{H}_2\text{O}_2$  was also conducted, including kerosene drop-flow testing, and fuel additive selection (candidate catalysts included manganese sulfate, copper sulfate, sodium nitroprusside, and potassium cuprocyanide mixed with hydrated hydrazine).<sup>17</sup> Satisfactory results were not achieved, but this exploration has been continuing, for example in Russia, where similar research on nontoxic, hypergolic propellants is being pursued. In 1993, a research program on additives to kerosene that can self-ignite with  $\text{H}_2\text{O}_2$  was initiated and test-firings on a low-thrust test thrust chamber were successfully conducted in China. This program is discussed in Sec. III of this paper.

### C. Cryogenic Bipropellants

Oxygen and hydrogen have been proposed for a number of reusable launch vehicles for on-orbit propulsion, primarily be-

cause of the ability to integrate the OMS/RCS with the main propulsion system and the high performance of 360 s for RCS and 460 s for OMS. Because of the low pressures typical of liquid hydrogen storage, the RCS will require pumps. Because of the low density (0.075 g/cc, 4.7 lbm/ft<sup>3</sup>) and low temperature of liquid hydrogen at  $-253^\circ\text{C}$  ( $-423^\circ\text{F}$ ), it is impractical to pressure-feed the propellants. A review of the technology has been presented by Reed and Schneider.<sup>18</sup> Many of the proposed systems have converted the hydrogen to a gas before the thruster, although a pulsing  $\text{O}_2/\text{H}_2$  thruster was demonstrated by Herr and Schoenman.<sup>19</sup> For low total impulse requirements, the propellants could be stored as high-pressure gas, given adequate space within the spacecraft. Methane provides an alternative to hydrogen. It provides a higher liquid storage density of 0.415 g/cc (26 lbm/ft<sup>3</sup>). It may also be possible to utilize a pressure-fed system for oxygen and methane, although the mass of the pressurant tanks may be an issue. Storage of the helium inside the propellant tanks at cryogenic temperatures would help eliminate this issue.

LOX and alcohol or kerosene provide a slightly higher specific impulse and a lower overall density as compared to MMH/NTO. Specific impulse ranges from 270 to 292 s for RCS, and 320 to 334 s for OMS. As a fuel, ethanol offers a low freezing point at  $-114^\circ\text{C}$  ( $-173^\circ\text{F}$ ), which will minimize freezing concerns in the injector and offers very clean combustion. Material compatibility is very good for ethanol and oxygen. However, oxygen at pressures above 3.4 MPa (500 psia) could ignite some stainless steels under particle impact. Because a pressure-fed system operates at less than 2.4 MPa (350 psia), this is less of an issue; however, proper material selection, system and component design, and system cleanliness are still critical to safety.<sup>20</sup> Kerosene offers higher performance, but produces excessive carbon combustion products and has a lower freezing point. There have also been some reports of explosive gels forming with the LOX. For these reasons, ethanol is a recommended propellant for attitude control systems. For RCS engines, the pulse mode spark ignition of gaseous oxygen and ethanol has been demonstrated in a 2.7 kN (620 lbf) engine by Lawver.<sup>21</sup>

### III. $\text{H}_2\text{O}_2$ and Kerosene, Additives, and Self-Ignition

The search for additives to kerosene to make it self-ignitable with  $\text{H}_2\text{O}_2$  is of great significance. The work presented here was conducted in China by the authors at the Shanghai Academy of Spaceflight Technology. The approach is to conduct additive screening tests<sup>22</sup> and then conduct hot-fire tests using a thrust chamber.

An ideal kerosene additive must have good ignition characteristics and good solubility in kerosene. In addition, it should have good chemical stability and be unable to ignite and burn in atmosphere, to ensure handling safety. The propellant combinations selected in this program are 90 and 96% concentration  $\text{H}_2\text{O}_2$  and China no. 3 aviation kerosene.

The additives are the salt of transition metals and combustible organic compounds, the former being the catalyst while the latter is the promoter. The catalyst, the promoter, and the complex formed by them must all be soluble in kerosene to make the fuel homogeneous. For the catalysts, the fuel-soluble metallic salts such as hydrocarbon sulfonates and carboxylates, e.g., naphthenates, naphthyl sulfonates, and dialkyl naphthyl sulfonates, with the cupric, ferrous, ferric, chromic, manganous, and cobaltous, etc., are chosen. For exploratory research, the amines, hydrazine, phenol, and aldehyde were selected as the promoters.

Ferrous naphthyl sulfonate was obtained through an exchange reaction of sodium naphthyl sulfonate and ferrous sulfate and extracted by alcohol and then refined by evaporating the solvent. Ferrous dibutyl naphthyl sulfonate was obtained through an exchange reaction of Nekal BX and salt of ferrous sulfate, extraction, and refinement, with the solvent removed

by vacuum flash distillation. Cupric dodecyl benzene sulfonate was obtained through an exchange reaction of sodium dodecyl benzene sulfonate and copper sulfite, extraction, and refinement, with the solvent removed by vacuum flash distillation.

Other organic acid salts of iron, copper, cobalt, chromium, and manganese were obtained through exchange reactions of sodium salts of organic acid and corresponding sulfates, extraction, and refinement, with the solvent removed by vacuum flash distillation. The promoters, such as phenol, aldehyde, amine, and hydrated hydrazine, are all commercially available chemicals.

#### A. $H_2O_2$ /Kerosene Additive Screening Tests

A drop-testing method is used in the ignition testing program. A number of drops of kerosene are dropped into a clean, dry porcelain crucible. Then the catalyst and promoter with a given composition are well mixed to make the mixture as homogeneous as possible. The dense  $H_2O_2$  is then dropped in and the reaction phenomena are observed.

Drop tests were made with different combinations of candidate catalysts and promoters. The tested catalysts include naphthalene sulfonates of iron, copper, cobalt, chromium, and manganese. The tested promoters include tert-butyl *p*-phenol, *p*-tert-butylcatechol, tetramethyl ethylene diamine, triethyl amine, hydrated hydrazine, unsymmetrical dimethyl hydrazine (UDMH), and crotonaldehyde. The test results show that all of these salts are insoluble in kerosene, and that the mixtures with the previously listed promoters are nonhomogeneous.

Although the naphthyl sulphonates, especially ferrous naphthyl sulfonate, if associated with promoters, can induce self-ignition and burning of kerosene with  $H_2O_2$ , their solubility in fuel is poor and cannot provide a homogeneous system, thus making them inappropriate for practical applications. For this reason, ferrous dibutyl naphthyl sulfonate will be used as a catalyst to improve solubility. The test results showed, however, that although this catalyst is soluble in kerosene, it can only catalyze the decomposition of  $H_2O_2$ . Self-ignition is poor, even with a promoter. For example, in ignition testing, the fuel-soluble additive formed by dibutyl naphthyl sulfonate with tetramethyl ethylene diamine first caused decomposition of  $H_2O_2$ , then ignition and burning.

More detailed research was conducted on the long carbon-chain dodecyl benzene sulfonate in an attempt to find a usable

catalyst with both good solubility and satisfactory ignition characteristics. The test results are shown in Table 4.

The solubility and flammability of the dodecyl benzene sulfonates are better than those of the naphthyl benzenes. Among them, the flammability of the ferrous dodecyl benzene sulfonate is by far the best. It can cause self-ignition, even without any promoter.

The phenol promoters, when mixed with the salts of iron and copper, will form related complexes. The phenols had the best flammability with the ferrous salts. The testing showed that the flammability and solubility of phenol, tert-butyl *p*-phenol, *p*-tert-butylcatechol, and pyrocatechol, are quite similar. Comparatively, the flammability of the *p*-tert-butylcatechol is much better. It was found in experiments that more of the phenol promoter is required than of the catalyst. Thus, to obtain satisfactory flammability, more additive is required in the kerosene.

The organic amines and hydrazine, when associated with the catalysts, also show a certain promoting action. The additives formed by the tetramethyl ethylene diamine or triethylamine and the salts of chromium or cobalt are soluble in kerosene, but the additives formed by tetramethyl ethylene diamine and triethylamine and the salts of copper, chromium, and iron can make the kerosene inflammable. The additive tetramethyl ethylene diamine/copper salt is only partially soluble in kerosene, but the flammability is very good. To obtain the best additive formation for further improving the self-ignitability and reducing the required additive quantity, tests on binary and ternary mixed salts were performed. The test results of this research are shown in Table 5.

In summary, the additive system of mixed salts of copper and cobalt/tetramethyl ethylene diamine is most attractive for instant ignition, stable combustion, and little required additive. Although they are partially soluble in kerosene, the addition of a small amount of solvent can obtain homogeneous liquid fuel. Alcohols are selected as the additive solvent for good solubility and self-flammability. Through extensive testing a satisfactory formation of kerosene additive with good solubility and flammability was found. The aforementioned tests were performed mainly with 80%  $H_2O_2$ . The results of a few tests with 95%  $H_2O_2$  showed that self-flammability of the propellant combination was even better, so that the amount of the required additive could be reduced.

Table 4 Initial screening of catalysts and promoters

Catalyst	Promotor	Solubility in kerosene	Ignition test result
Cobalt dodecyl benzene sulfonate	Non	Partly soluble	Gentle decomposition
	Tert-butyl <i>p</i> -phenol	Soluble	Gentle decomposition
	<i>p</i> -tert-butyl catechol	Soluble	Gentle decomposition
	Tetramethyl ethylene diamine	Soluble, with green complex	Large amount of bubbles with sparks
	Triethyl amine	Soluble, with green complex	Violent decomposition with foam
	UDMH	Soluble	Foaming then igniting
Chromium dodecyl benzene sulfonate	Non	Partly soluble	No noticeable reaction
	Tert-butyl <i>p</i> -phenol	Basically soluble	No noticeable reaction
	<i>p</i> -tert-butylcatechol	Soluble	Fluid becomes brown
	Tetramethyl ethylene diamine	Soluble with yellow complex	Bubbling, igniting in 5 seconds
	Triethyl amine	Soluble with dark-green complex	Fluid becomes brown
	UDMH	Insoluble	Igniting in 2 s
Copper dodecyl benzene sulfonate	Non	Soluble	No noticeable reaction
	Tert-butyl <i>p</i> -phenol	Soluble with yellow-green complex	Gentle decomposition
	<i>p</i> -tert-butylcatechol	Soluble with yellow complex	Gentle decomposition
	Tetramethyl ethylene diamine	Partly soluble, dark-blue complex	Immediate ignition
	Triethyl amine	Insoluble, blue complex formation	Gentle decomposition with bubbles
	UDMH	Insoluble, brown complex formation	Violent decomposition
Ferrous dodecyl benzene sulfonate	Non	Soluble	Ignition with 8 s lag
	Tert-butyl <i>p</i> -phenol	Soluble, dark-green complex	Immediate ignition
	<i>p</i> -tert-butylcatechol	Soluble, dark-green complex	Immediate ignition
	Tetramethyl ethylene diamine	Insoluble, yellow-green complex	Bubbling first, ignition in 2–3 s with blasting
	Triethyl amine	Insoluble, yellow-green complex	Large amount of bubbles
	UDMH	Insoluble, brown-yellow complex	Large amount of bubbles

Table 5 Final evaluation of catalysts and promoters

Catalyst	Promotor	Solubility in kerosene	Test result
Mixed salt of iron and copper	<i>p</i> -tert-butylcatechol	Soluble, with green complex	Instant ignition
Mixed salt of iron and copper	UDMH	Soluble, with yellow complex	Instant ignition, flame stable
Mixed salt of copper and cobalt	Tetramethyl ethylene diamine	Partially soluble, dark-blue solution	Instant ignition with some white foam
Mixed salt of copper and chromium	Tetramethyl ethylene diamine	Partially soluble, dark-blue solution	Instant ignition with some white foam
Mixed salt of copper, cobalt, and chromium	Tetramethyl ethylene diamine	Partially soluble, dark-blue solution	Instant ignition with some white foam

### B. Test Firings of H<sub>2</sub>O<sub>2</sub>/Kerosene

Based on the self-ignition research on additives for the kerosene and H<sub>2</sub>O<sub>2</sub> performed in the laboratory, a test-firing program was initiated on a specially designed thrust chamber to evaluate the self-ignition and steady-state combustion characteristics of the propellant combination.

In addition to the additives, the self-ignition performance is closely related to injection, atomization, and mixing.<sup>23</sup> Thus, a thrust chamber with two different types of injectors was designed. The type A injector is composed of a central swirler fuel element and a peripheral ring of 24 self-impinging doublet oxidizer elements, whereas the Type B is composed of a central swirler fuel element and a peripheral ring of six swirler oxidizer elements.

Because the purpose of the tests is to evaluate self-ignition characteristics, perfect performance was not pursued and the structure of the thrust chamber was designed to be as simple as possible. A graphite liner is incorporated in the body, on which a phenolic-resin-dipped glass fiber tape is wrapped for thermal insulation. The closeout of the chamber is made of stainless steel and is bolted to the injector. The oxidizer and fuel feed lines are controlled by solenoid valves for propellant entry sequence control and for pulse-mode testing to be conducted in the future.

The main design characteristics of the test-piece thrust chamber are shown next. Vacuum thrust ( $F_v$ ) 200 N, chamber pressure ( $P_c$ ) 0.7 MPa, mixture ratio (MR) 7.0, chamber diameter ( $D_c$ ) 58 mm, throat diameter ( $D_t$ ) 16 mm, nozzle exit diameter ( $D_e$ ) 32 mm, chamber contraction ratio ( $c$ ) 13.4, nozzle area ratio ( $e$ ) 4, and chamber characteristic length  $L^*$  1.5 m. The propellant combination used for the testing is 94–96% concentration H<sub>2</sub>O<sub>2</sub>; China no. 3 aviation kerosene and additive.

Since 1996, four test-firings have been conducted: two in the type A injector and two in the type B injector. During the second test-firing of the type A injector, an explosion of the injector occurred because of incomplete purging of the injector manifold. The other three test-firings were successful with normal startup, stable combustion, bright exhaust, and normal shutdown.

Disassembly after test-firings showed that the thrust chamber was always in good conditions. The steady-state parameters of the test-firings are listed in Table 6. The startup and ignition were normal, realizing the self-ignition of H<sub>2</sub>O<sub>2</sub>/kerosene (additive blended), just as the hypergolic propellant NTO/MMH. The fuel lead sequence was used for all test-firings. No excessive pressure peaks were identified, which means that the startup sequence is feasible. The nominal mixture ratio of this propellant combination is about 7–7.5. The lack of experience in system tuning led to a much higher or lower test-firing mixture ratio, as shown in Table 6. However, this propellant combination can be self-ignitable in the mixture ratio range from 4.25 to 10.49 with a smooth startup, which further demonstrates the feasibility of utilizing hydrogen peroxide and additive-added kerosene to obtain a self-ignitable propellant combination. The observed steady-state performance is not high, with a combustion efficiency of about 0.84. This results from the mixture ratio being either too high or too low, and high heat sink losses caused by large thermal capac-

Table 6 Measured performance

Number	$P_c$ MPa	$P_{\infty}$ MPa	$P_{cst}$ MPa	$q_{cst}$ g/s	$q_{\infty}$ g/s	MR	Test time, s
B101	0.617	0.936	1.483	8.57	89.9	10.49	15
B102	0.407	0.749	2.524	16.49	68.63	4.245	20

itance of the chamber. A comparison of the test-firing results on types A and B injectors shows that the combustion efficiency is higher for type B than for type A.

Several conclusions are drawn from the research. The percentage of the promoter in the present research for kerosene additives is less than 5%. However, these promoters are slightly toxic, thus the additive-added kerosene is, strictly speaking, a low-toxicity propellant. An extensive research program should be conducted to develop nontoxic additives or to reduce the percentage of toxic additives to an allowable level. A research program on additives without a promoter is now being pursued. As far as the propellant itself is concerned, further work on improving the solubility, stability in long-time storage, ignition time lag, etc., should be performed. The thrust chamber performance including start sequence, start acceleration, steady-state combustion performance, pulse-mode operation performance, and high-altitude performance, etc., must be improved. The propulsion system scheme for spacecraft attitude/orbit control including system design, propellant tanks, and propellant management devices design, and long-life, high-reliability, quick-response solenoid valves design, etc., must be optimized.

Based on the results from this research work, it can be expected that the propellant combination of H<sub>2</sub>/O<sub>2</sub> and additive-added kerosene will be used for spacecraft on-orbit propulsion systems in the future and consequently realize the innovation of a nontoxic, self-ignitable spacecraft on-orbit propulsion system.

## IV. LO<sub>2</sub>/Ethanol System for Space Shuttle Orbiter

A LOX system, with either alcohol or hydrocarbon fuel, provides high performance commonality with other spacecraft systems, and supports future exploration missions. The rationale and tradeoffs involved in the selection of LOX and alcohol for upgrades to the Orbiter is discussed in more detail by Hurlbert and Moreland.<sup>24</sup> They also discuss tradeoffs concerning the RCS feed system and pressure-fed vs pump-fed systems. This paper will focus on a pressure-fed subcritical LOX system. The system can integrate the auxiliary power unit (APU) and eliminate the hydrazine system. The ethanol may be used to cool the APU lube oil and the hydraulic system and eliminate the water spray boilers. The LOX can also be used to eliminate the ammonia boilers that cool the active thermal control system under high heat loads. The following is a description of the requirements and a conceptual design.

### A. Cryogenic Propellant Storage System

The existing Space Shuttle Orbiter OMS/RCS uses three different tank configurations to meet the Orbiter operational requirements and provide gas-free propellant over all mission

phases. Each tank configuration is designed to perform optimally over distinct operational envelopes, resulting in a tank system that provides relatively high performance over the full Orbiter operational envelope. The forward RCS tank is designed for efficient low-*g* acquisition under acceleration environments less than 0.05 *g*. The aft RCS tank design, while designed for low-*g* acquisition, also incorporates a collector and sump to take advantage of bulk propellant positioning under known entry accelerations. Finally, the OMS tank is designed to store the majority of the propellant and to support high propellant outflow rates for translation, deorbit, and dump burns in addition to limited RCS usage in an interconnect mode. The OMS tank is not a low-*g* tank and relies on propellant settling from burn accelerations to aid in efficient acquisition.

For the nontoxic OMS/RCS Orbiter upgrade, it was desired to develop a single tank configuration that could operate efficiently over the full operational envelope. Having just one tank configuration would reduce the total vehicle tank count from 10 (two forward RCS, four aft RCS, and four OMS) to four (four OMS/RCS), providing cost savings by eliminating the need for multiple tank design, development, and certification programs. The reduced tankage also results in the added benefit of improved thermal efficiency, which is critical for LOX storage, by reducing surface area and penetration heat leaks. A LOX tank conceptual design capable of meeting the Orbiter operational requirements has been defined. The tank is sized for 4500 kg (10,000 lbm) of LOX, which will be loaded in parallel with the main propulsion external tank prior to launch. The oxygen will be loaded at 90 K (162°R), which is saturation temperature at 0.1 MPa (14.7 psia).

The tank will require a pressure control system to destratify and maintain the bulk propellant temperature below 102 K (185°R) at the end of the mission. Pressure control concepts that are being considered include axial jet and spray-bar mixer configurations, both of which incorporate a thermodynamic vent system. With the tank operating at 2.3 MPa (350 psia), sufficient subcooling of approximately 33 K (60°R) will be maintained to preclude going two-phase when feed system and engine pressure drops are encountered.

High accuracy, zero-*g* mass gauging remains a challenge for cryogenic systems. The compression mass gauge (CMG) concept shows potential for being able to provide mass gauging at an accuracy approaching 1%. The CMG utilizes a bellows to produce a known small volume change within the tank, resulting in ullage compression and a corresponding pressure change. Based on the measured pressure change and the em-

pirical determination of key fluid and thermodynamic coefficients, the liquid quantity in the tank can be determined. Alternate concepts for mass gauging, *P-V-T* determination, flow integration, liquid level sensors, etc., are also being investigated for possible use.

While this design focuses on the LOX tank, the same basic design should carry over to the ethanol tank. The ethanol tank may be of reduced length but common diameter. It does not require a pressure-control system, but it may require changes in the screen mesh and channel cross section of the liquid acquisition device to optimize for the different fluid properties. A tradeoff study of various tank configurations and acquisition concepts resulted in a possible configuration as shown in Fig. 2. On-going tradeoff and design definition studies are evaluating variations of, and in some cases, very different design concepts for the various tank subsystems to meet the overall performance goals.

One design concept is a hybrid composite tank pressure vessel, incorporating an Inconel 718 shell with a composite overwrap of the barrel section only, providing weight benefits over an all metal tank while keeping costs low as compared to a full composite tank.

The tank will require insulation to protect the 103 K (185°R) end-of-mission requirement for a nominal 7-day mission. This goal can be achieved if the on-orbit heat flux into the tank can be limited to approximately 0.16 W/m<sup>2</sup> (0.5 Btu/ft<sup>2</sup>/h), which should provide sufficient margin to keep the overall heat leak below an effective flux of 0.32 W/m<sup>2</sup> (1.0 Btu/ft<sup>2</sup>/h) once penetration heat leaks and other inefficiencies are accounted for.

One possible insulation configuration involves approximately 1.9 cm ( $\frac{3}{4}$  in.) of Isofoam SS-1171 on the pressure vessel, followed by the varying density application of 10 layers of double-aluminized Mylar using an innovative commercial spiral wrap technique. The tank assembly would then be covered by an outer protective shell of Nomex or other durable material. The foam prevents icing on the ground and limits ground hold boil-off to approximately 3% over a 2-h period, which can be easily made up through the ground replenish loop. Condensation of moisture within the insulation system during the prelaunch, ascent, and entry phases must be addressed to ensure no adverse effects on insulation performance or life, and to preclude the need for extensive ground processing to dry out the insulation. Use of active purging and/or alternate insulation materials insensitive to moisture are being considered as possible options.

The acquisition system concept is a hybrid of the existing OMS and RCS tanks, incorporating unique design features

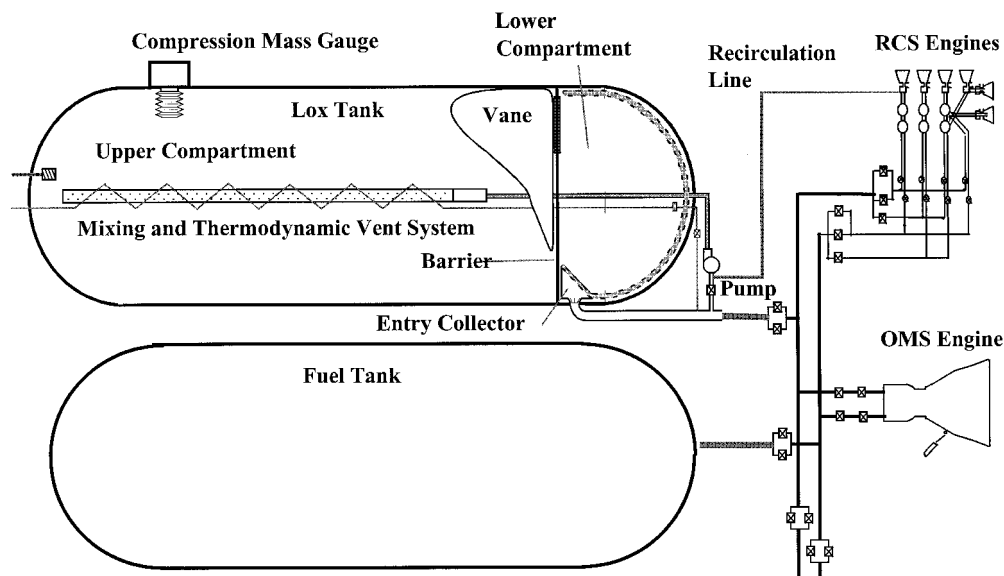


Fig. 2 Layout of propellant system for Space Shuttle on-orbit propulsion system.



from each. The tank is compartmented with a solid barrier that provides communication between the upper and lower compartments through a plenum screen. The upper compartment incorporates a vane system to assist in maintaining liquid propellant over the plenum during low-g, on-orbit operations. The lower compartment incorporates a surface tension screen channel device with an entry sump. The tank is designed to accommodate limited gas transfer into the lower compartment, primarily during the OMS engine startup and on-orbit RCS usage when propellant positioning over the plenum cannot be guaranteed. The lower compartment volume is sized to accommodate this gas transfer as well as the return-to-launch-site (RTLS) abort dump, which would severely stress the screen acquisition system. Typical operation of the acquisition system over multiple mission phases is described by the picture sequences shown in Fig. 3.

Propellant loading must be performed in a manner that entirely fills the lower compartment with liquid propellant while minimizing the amount of gas trapped in the lower compart-

ment channels and below the barrier. Because premature screen wet over during the fill cannot be precluded, bleed lines would probably be required to bleed off any trapped gas in the channels during the fill procedure. The propellant remains settled in the tank during loading and launch as shown under accelerations varying from 1 to 3 g (Fig. 3, sequence 1). The initial tank demand on a nominal mission would occur at mated coast and external tank (ET) separation following the main engine cutoff. Because of the transient acceleration environment during this mission phase, liquid propellant cannot be assured at the plenum and a limited amount of gas transfer to the lower compartment, while unlikely, may occur during the RCS operation (sequence 2). The OMS circularization burn, as well as all subsequent OMS burns, could result in additional gas transfer to the lower compartment at initial engine startup, until the acceleration positions the bulk propellant over the plenum (sequence 3). On-orbit primary RCS usage for attitude control and small-velocity change maneuvers could result in additional gas transfer to the lower compartment. The

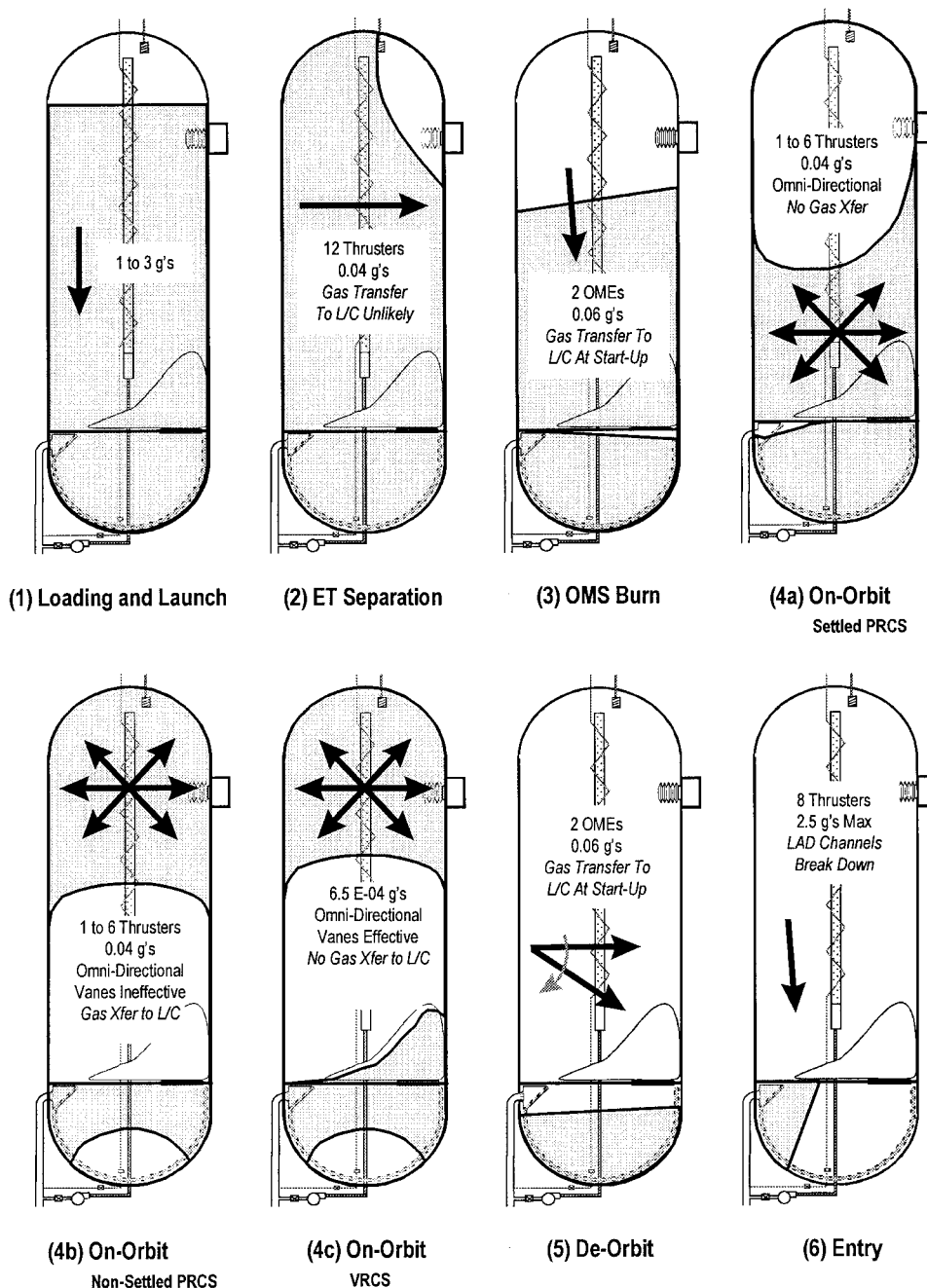


Fig. 3 Representative bulk fluid level and position at various phases of mission.

degree of gas transferred would depend on the positioning of the bulk propellant in the upper compartment relative to the plenum at the time the burns take place (sequences 4a and 4b). The upper compartment vane system is designed to maintain liquid over the plenum at acceleration levels typical of vernier RCS usage for attitude control, precluding gas transfer to the lower compartment (sequence 4c). The lower compartment screen acquisition system is designed to operate at the maximum outflow rate without exceeding its design bubble point, assuming the worst case quantity of gas has been transferred beneath the barrier. The lower compartment screen channels maintain full integrity through the OMS deorbit burn, which results in the depletion of liquid propellant from the upper compartment (sequence 5). The screen acquisition system eventually breaks down during the high accelerations experienced during entry, but will continue to operate at high efficiency, because the entry acceleration positions the propellant over the entry sump (sequence 6). Following landing, the residual propellant remains positioned over the collector, providing access to the residual liquid oxygen for use in the LO<sub>2</sub> boiler to support Orbiter system cooling (not shown).

## B. Propellant Feed System

The propellant feed system must store and supply oxidizer and fuel to the RCS thrusters and OMS engines in a consistent and reliable manner to have a successful mission, and to ensure that performance characteristics of the thrusters/engines do not fluctuate as a result of varying MR. The OMS engine will need to produce between 17.8 and 27 kN (4000 to 6000 lbf) of thrust, whereas the RCS thruster needs to produce 3.9 kN (870 lbf). For an OMS engine producing 27 kN at a chamber pressure of 0.86 MPa (125 psia) with a nozzle expansion ratio of 55:1, the required oxidizer flow rate to fire both OMS engines is 10.8 kg/s (23.8 lbm/s), and the ethanol flow rate is 6 kg/s (13.2 lbm/s). These flow rates are for an engine  $I_{sp}$  equal to 323 s with an MR = 1.8. The RCS thrusters are envisioned to be dual thrust, with the igniter operating by itself to produce a thrust level of 110 N (25 lbf), whereas operation of the main propellant valves produces 3.9-kN (870-lbf) thrust. For a combustion chamber pressure of 1 MPa (150 psia), with a nozzle expansion ratio of 22:1, a reasonable  $I_{sp}$  of 270 to 290 s can be expected. For a mixture ratio of 1.8, this requires an oxidizer mass flow rate of 0.9 kg/s (2.0 lbm/s) supplied to each thruster. For external tank separation, simultaneous firings of up to 12 thrusters is required. The feed system must be capable of this flow rate for external tank separation and all other mission phases.

Other mission phases that require high propellant usage rates are the OMS engine burns for orbit insertion and deorbit, backup RCS engine deorbit, and RCS thruster burns for a single engine roll control (SERC) abort scenario. OMS engine burns consume approximately 26% of the propellants in 560 s, whereas the ET separation maneuver requires 88 kg (194 lbm) of oxidizer in 10 s and SERC requires 454 kg (1000 lbm) of oxidizer in 65 s.

A gaseous oxygen (GOX) RCS feed system could use a high-pressure storage bottle to supplement the flow and meet high demands for oxygen during ET separation, SERC, and entry. The lower flow demands can be met by a heat exchanger with the active thermal control system. This system, while practical, has some drawbacks, such as 1) the need to load GOX, 2) the use of Monel materials in the high-pressure system, 3) the lack of RCS backup deorbit capability, and 4) the increased mass of the tanks. None of these issues are insurmountable. The backup deorbit function could be provided by a special RCS engine that runs on LOX in the +X location. The deorbit burn times are up to 800 s for the RCS engine. Some form of temperature compensation during the blowdown of the oxygen tank is also required to maintain the mixture ratio.

A cryogenic RCS feed system option for Shuttle upgrade is a system that stores LOX in the feedlines at all times and feeds

subcritical oxygen directly to the RCS thrusters. This concept was first considered to be possible by Orton et al.<sup>25</sup> The advantage of this feedsystem concept is the inherent simplicity in terms of the number of fluid components and volume requirements as compared to a GOX feedsystem, which has large storage volume requirements and requires a gasification system. A cryogenic liquid distribution system was also proposed by Akkerman<sup>26</sup> (which distributed the O<sub>2</sub>/H<sub>2</sub> at supercritical pressures).

The challenge of such a system is to thermally isolate the LOX from the environment for the duration of the mission, maintain the cryogen in a subcooled state without active cooling systems, and minimize oxygen loss caused by boiloff. The technical challenges of the RCS cryogenic feedline are the definition of the sources of heat leak into the system, determination of the most effective way to thermally insulate the feedline and LOX tank, and the determination of the most efficient ways to account for the heat leak into the system. Of course, no system will be perfect and totally eliminate heat leak into the system, but the objective is to determine the most effective insulation in terms of low thermal conductivity, light weight, and ease of installation and maintenance. Potential solutions are 1) the use of recirculation pumps to thermally condition the feedline, 2) operational procedures such as strategically firing thrusters to circulate the LOX, and 3) vapor-cooling of the lines. Other technical challenges are the design of a LOX ethanol thruster that minimizes vaporization of the LOX at startup and minimizes the heat leak back into the LOX at the propellant valve. These issues and others are currently being studied and evaluated in support of the RCS cryogenic feedline option.

The heat leak into the LOX feedsystem in the Shuttle environment can be separated into two categories, the ambient heat leak and thruster induced heat leak. Ambient heat leak is attributed to heat leak from the Shuttle environment into the feedlines and propellant tank. Ambient heat leak is a combination of conductive heat leak through the structural tie-downs of the feedlines and structural support of the LOX propellant storage tanks and the radiative heat leak into the system. Thruster-induced heat leak is the conductive heat leak from the high-temperature injector and nozzle assembly of a recently fired thruster and also the heat flux imposed on the injector face because of direct impingement of solar flux on a sun-facing thruster.

The current system configuration for the Shuttle RCS thrusters utilizes a ring manifold concept in which the four thrusters are fed in series off of one manifold. The rationale is that this configuration would improve circulation of fluids to thermally condition the feedlines. Instead of a recirculation pump at the end of each thruster, as required in a parallel-fed system, the ring manifold configuration only requires as many recirculation pumps as there are manifolds, yet provides recirculation flow to each of the thrusters. The feedline diameters must be increased to account for the increased flow rates and corresponding increase in pressure drop through the feedlines.

The feedlines must be insulated from the ambient and thruster-induced heat leaks. Traditional cryogenic insulations are being evaluated and these include spray-on foam insulations (SOFI), vacuum jacketed lines, multilayer insulations (MLI), and other relatively new types of insulators such as the aerogel-based insulations. Low conductivity, weight vs performance, durability, and installation and maintenance are all factors that must be considered when evaluating these insulations. Aerogel-based insulations are a promising candidate because of their low thermal conductivity at ambient pressure and in a vacuum environment. This is a result of the low density of the aerogel, which has low conductive heat transfer, and the use of radiation absorbing materials impregnated into the insulation to reduce radiative heat transfer.

Cryogenic storage systems designed in the past using combinations of MLI with other ambient pressure insulations have

proven to maintain a heat flux level of less than  $0.32 \text{ W/m}^2$  ( $1.0 \text{ Btu/ft}^2/\text{h}$ ) into the cryogen. The challenge for the LOX RCS feedline option is that insulating the feedlines with numerous bends, fittings, and complex shape fluid components will be much more difficult than insulating the geometrically simple, uniform surface area of a large storage tank. Also, the surface area per volume ratio for the feedline system is much larger than for cryogen storage tanks, so that the heat flux per volume will be higher in feedlines than in traditional cryogen storage tanks. The conservative goal for heat flux levels into the LOX storage tanks will be approximately  $0.32 \text{ W/m}^2$  ( $1.0 \text{ Btu/ft}^2/\text{h}$ ) into the tank, and a heat flux of  $0.64 \text{ W/m}^2$  ( $2.0 \text{ Btu/ft}^2/\text{h}$ ) into the feedline system. Recirculation pumps have been sized in a preliminary analysis effort to determine the required duty cycle and pump flow rates required to maintain zero-quality LOX in the feedlines at all times. A schematic of the model representing the propellant tank with a thermodynamic vent system (TVS), the main feedline from the aft to forward section, and manifold feed to the thrusters is shown in Fig. 2. The recirculation pump routes the LOX back into the LOX tank.

Thermal/fluid systems analysis results show that each of the four recirculation pumps in the forward section should be sized to provide a flow rate of at least  $0.18 \text{ kg/s}$  ( $0.39 \text{ lbm/s}$ ) and operate every  $5.3 \text{ h}$  to maintain the LOX temperature below  $103 \text{ K}$  ( $185^\circ\text{R}$ ). The allowable temperature range of  $90\text{--}103^\circ\text{K}$  ( $162\text{--}185^\circ\text{R}$ ) was chosen to limit the density variation to less than  $10\%$  to avoid drastic changes in MR to the thruster and consequent performance. Figure 4 shows the temperature of LOX as a function of time in a feedline for line diameters of  $2.5$  and  $5 \text{ cm}$  and for heat fluxes of  $3$ ,  $6$ , and  $9 \text{ W/m}^2$ . It may be possible to maintain temperatures below  $103 \text{ K}$  in a quiescent line for up to  $24 \text{ h}$ .

### C. RCS Engines

Two types of RCS engines are being evaluated, GOX and LOX. These RCS engines will have four major technical and cost drivers; valves, injector, chamber, and ignitor. One advantage of a LOX thruster is that the valve size is much smaller. A GOX valve for an RCS engine is rather large. The RCS envelope can be expanded to allow the larger valve, but this will require structural interface changes. For the Shuttle RCS, the main drivers for the valves are reliability (leakage) and cost. To allow a directing acting design, new valve requirements for the Orbiter upgrade will allow a longer opening

time of  $50 \text{ ms}$  vs the current  $30 \text{ ms}$  and also allow a higher peak current of  $3 \text{ A}$ . This should reduce the piece-parts from approximately 114 for the current pilot operated valve to fewer than 20 and, hence, reduce the cost. Valve leakage will also be improved because of less corrosive propellant and the elimination of the pilot stage as a second leak path.

A concern with LOX RCS thrusters is the question of the quality of LOX supplied to the combustion chamber at the initial startup of the thruster, when the temperature of the injector assembly is above the vaporization temperature of the LOX. Directly related to this question is how much time will elapse before zero-quality oxygen makes it way into the combustion chamber. The quality and length of time for two-phase oxygen entering the combustion chamber determines the mixture ratio, and this ultimately determines the thrust level and performance of the thruster. One of the objectives of a LOX thruster design will be to minimize the conduction of heat through the injector assembly and the valve plate. The requirement for the peak heat soak-back will be approximately  $0.59 \text{ W/m}^2$  ( $2.0 \text{ Btu/h}$ ). This appears possible assuming a  $2.5 \text{ cm}$  ( $1 \text{ in.}$ ) long thermal standoff using Inconel 718. Another possible design uses the oxygen to cool the chamber and to gasify the LOX before it gets to the injector. In terms of the injector, GOX presents some difficulty because of the low density and high injection velocities.

The chamber is another cost driver. Current RCS engines use columbium chambers with a disilicide coating. This is costly from the standpoint of both manufacturing and of operation because of the checkout of the coating. From a systems perspective, the RCS engine performance is not as critical as the OMS engine performance because the RCS uses about one-fourth the total amount of propellant or less. If a reduction in performance is traded for a reduction in wall temperature, then lower-cost materials, such as a Haynes alloy, can be used.

Two primary options for the ignition device are being explored: spark ignition and laser ignition. The ignition of the main chamber may be done using a torch. The torch will serve two purposes: 1) to control the propellant conditions more precisely at the point of ignition, and 2) to function as a vernier in a dual-thrust RCS engine. For the Orbiter, this means that the vehicle will have a two-fault tolerant Vernier RCS system of 38 thrusters rather than the current zero-fault tolerant system of six thrusters. This is significant in that the verniers are critical for missions such as certain rendezvous scenarios and reboost of a space station. The ignitor for LOX will be required

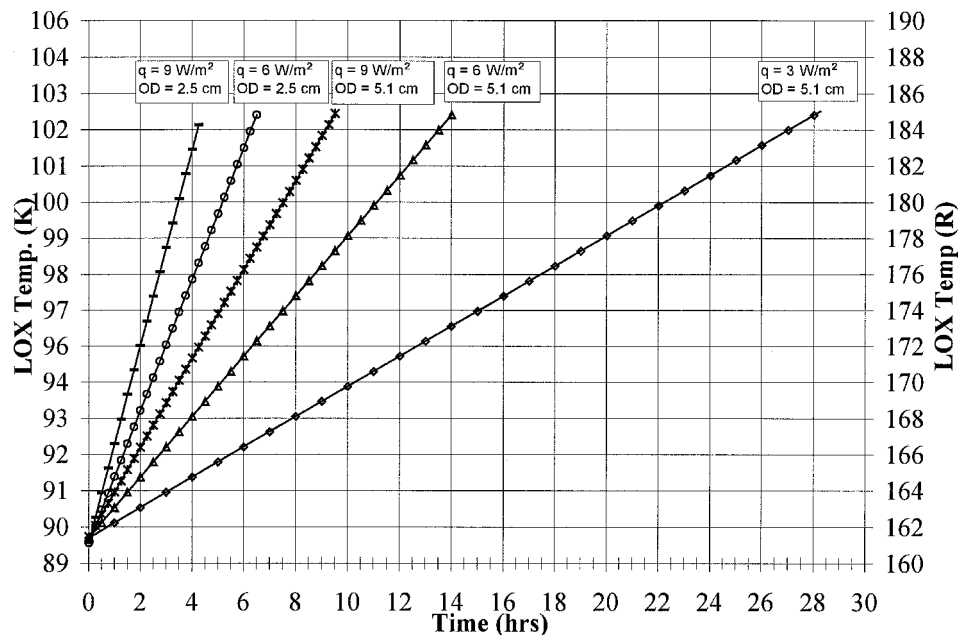


Fig. 4 LOX temperatures in feedlines vs time for various levels of heat leak into system.

to ignite over a range of propellant conditions from liquid to gas. This will make the RCS engine more tolerant of poor-quality oxygen at the inlet valve, hence, easier to interface with the system.

## V. Conclusions

Nontoxic propellants are available to vehicle and propulsion system designers to replace toxic Earth-storable propellants. Hydrogen peroxide and ethanol can support long-duration missions for many spacecraft. Significant progress in making these propellants hypergolic has been made. For short-duration missions like the Space Shuttle Orbiter, a cryogenic liquid RCS system may even be simple and practical, and provide propulsion systems capable of using propellant manufactured throughout the solar system. Technology development is underway to examine LOX and ethanol for Orbiter OMS/RCS. HAN-based monopropellants may eventually provide the replacement for hydrazine for long-duration expendable missions. The operational benefits of low-cost, nontoxic, noncorrosive propellants will provide a firm foundation for the continued exploration and development of space.

## Acknowledgment

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## References

- <sup>1</sup>Sanders, J. B., "Integrated Propulsion and ISRU Propellant Production System for Mars Sample Return," AIAA Paper 95-2641, July 1995.
- <sup>2</sup>Barber, H. E., "Advanced Pressurization Systems Technology Program," TR Air Force Rocket Propulsion Lab., TR-66-278, Nov. 1966.
- <sup>3</sup>Barber, H. E., Flakenstein, G. L., Buell, C. A., and Gurnitz, R. N., "Microthrusters Employing Catalytically Reacted  $N_2O_2$ - $H_2$  Gas Mixtures, Tridyne," *Journal of Spacecraft and Rockets*, Vol. 8, No. 2, 1971, pp. 111-116.
- <sup>4</sup>Mueller, J. M., and McFarlane, J. S., "Design of Tridyne Pressurization System for Liquid Oxygen Polybutadiene Hybrid Rocket Motors," AIAA Paper 91-2406, June 1991.
- <sup>5</sup>French, J. R., and Crisalli, D. E., "Minature Warm Gas Thruster Development," Final Rept., NAS3-96035, Feb. 1997.
- <sup>6</sup>*Liquid Propellant XM1846 Handbook*, Jet Propulsion Lab., U.S. Department of the Army, Armament Research, Development, and Engineering Center, Picatinny Arsenal, NJ, July 1994.
- <sup>7</sup>Jankovsky, R. S., "HAN-Based Monopropellant Assessment for Spacecraft," NASA TM-107287 and AIAA Paper 96-2863, July 1996.
- <sup>8</sup>Jankovsky, R. S., and Oleson, S. R., "HAN-Based Monopropellant Propulsion System with Applications," NASA TM-107407, Jan. 1997.
- <sup>9</sup>Meinhardt, D., Brewster, G., Christofferson, S., and Wuchere, E. J., "Development and Testing of New, HAN-Based Monopropellants in Small Rocket Thrusters," AIAA Paper 98-4006, July 1998.
- <sup>10</sup>Andrews, D., "Advantages of Hydrogen Peroxide as a Rocket Oxidant," *Journal of the British Interplanetary Society*, Vol. 43, July 1990, pp. 319-328.
- <sup>11</sup>Moser, D., "Low Cost Liquid Upper Stage for Small Launch Vehicles," AIAA Paper 94-3024, June 1994.
- <sup>12</sup>Wilkinson, C. L., and Bentz, M. D., "Reduced-Hazard Propellants for Auxiliary Propulsion," Boeing Aerospace and Electronics, International Aeronautical Federation, Seattle, WA, 90-252, Oct. 1990.
- <sup>13</sup>Clapp, M. D., and Hunter, M. W., "A Single Stage to Orbit Rocket with Non-Cryogenic Propellants," AIAA Paper 93-2285, June 1993.
- <sup>14</sup>Stone, H. W., "HL-20 Sub-System Design," *Journal of Spacecraft and Rockets*, Vol. 30, No. 5, 1993, pp. 590-596.
- <sup>15</sup>Cort, R. E., Riccio, J. R., and Sanders, J., "Non-Toxic On-Orbit Propulsion for Advanced Space Vehicle Applications," AIAA Paper 95-2974, July 1995.
- <sup>16</sup>Andrews, D., and Sunley, H., "The Gamma Rocket Engine for Black Knight," *Journal of the British Interplanetary Society*, Vol. 143, 1990, pp. 301-310.
- <sup>17</sup>Kelley, T. F., "Self-Ignition Fuels, Exploratory Work with  $H_2O_2$  as an Oxidant," AD89-6017, Oct. 1947.
- <sup>18</sup>Reed, B. D., and Schneider, S. J., "Hydrogen/Oxygen Auxiliary Propulsion Technology," NASA Lewis Research Center, TM-105249, Sept. 1991.
- <sup>19</sup>Herr, P. N., and Schoenman, L., "Demonstration of a Pulsing Liquid Hydrogen/Liquid Oxygen Thruster," AIAA Paper 73-1244, 1973.
- <sup>20</sup>"Safety Standard for Oxygen and Oxygen System," NASA Safety Standard NSS 1740.15, Jan. 1996.
- <sup>21</sup>Lawver, B. R., "Ignition Characterization of LOX/Hydrocarbon Propellants," NAS 9-16639, Aerojet, Sacramento, CA, April 1985.
- <sup>22</sup>Gu, Q., "Additives Research for Self-Ignition of  $H_2O_2$ /Kerosene Propellant Combination," Internal Rept., 801 Inst., Shanghai Academy of Spaceflight Technology, Shanghai, PRC, Sept. 1996.
- <sup>23</sup>Tan, W., "Design of the Test Thrust Chamber for  $H_2O_2$ /Kerosene Test Firing," Internal Rept., 801 Inst., Shanghai Academy of Spaceflight Technology, Shanghai, PRC, Feb. 1996.
- <sup>24</sup>Hurlbert, E. A., and Moreland, R., "A Non-Toxic Reusable On-Board Propulsion System for Orbiter Upgrade and the Human Exploration and Development of Space," Space Technology and Applications International Forum, American Inst. of Physics, CP 387, Albuquerque NM, Jan. 1997.
- <sup>25</sup>Orton, G. F., Mark, T. D., and Weber, D. D., "LOX/Hydrocarbon Auxiliary Propulsion System Study," NAS9-16305, MDC E2548, July 1982.
- <sup>26</sup>Akkerman, J., "Shuttle RCS Cryogenic Liquid Distribution System Study," NASA-MSD, TM X-68913, N73-16765, Jan. 1972.