JOURNAL OF PROPULSION AND POWER
Vol. 22, No. 6, November–December 2006

Overview of United States Space Propulsion Technology and Associated Space Transportation Systems

Robert L. Sackheim*
NASA Marshall Space Flight Center, Huntsville, Alabama 35812

DOI: 10.2514/1.23257

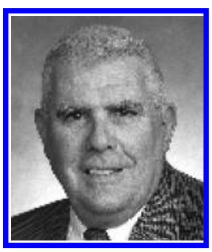
The space propulsion industry and, indeed, the overall space transportation industry in the United States have been in decline since the first human landing on the moon in 1969. The hoped-for reversal to this decline through the space shuttle and space station programs never really materialized. From an 80% market share in the late 1970s, the U.S. share of the world launch market fell to ∼20% by 2002. During that period, only two new booster engines have been developed and flight certified in this country. Only limited progress has been made in reducing engine costs or increasing performance although these factors are not necessarily directly related. Upper-stage and in-space propulsion in the United States have not fared much better in the world market. On the other hand, space-faring nations in Europe, the Middle East, Asia, and the former Soviet Union are believed to have developed 40–50 new, high-performance engines over the same period. This trend will have to be reversed to enable future exploration missions. The intent of this paper is to summarize past propulsion-system development, assess the current status of U.S. space propulsion, survey future options, evaluate potential impact of ultra low-cost, small launch-vehicle programs, and discuss some future propulsion needs for space exploration.

I. Introduction

THE U.S. rocket propulsion industry and associated space transportation business has been in a steady state of decline since the end of the Apollo era (∼1972), although the actual steep funding, and associated manpower, roll-off began immediately after the successful first human landing on the moon, Apollo 11, in 1969. A turnaround in the propulsion and space transportation industry was expected after the space shuttle, and even, ultimately, the space station, program was authorized to proceed, but the turnaround never materialized. The space shuttle program [or National Space Transportation System (NSTS)], which had to develop three new liquid rocket engines [space shuttle main engine (SSME), orbital maneuvering engine (OME), and reaction control engine (RCE)] and the world's first large, segmented, and reusable solid rocket motor did not reverse the decline from the Apollo era; it only slowed down the rate of decline until the late 1970s, as shown in Fig. 1.

In general, space transportation and rocket propulsion technology development in the United States for all aspects of space flight applications has significantly lagged behind the rest of the world since the initial flight certification of the space shuttle Space Transportation System (STS). This lack of progress in advancing rocket propulsion technologies over such a long period has resulted in several deficiencies in today's U.S. national space program. Most notable of these is the reduced reliability in U.S. launch and space vehicles, as evidenced by the increased number of flight failures during the late 1990s and into the new decade, as well as the large loss of U.S. market share in both the space launch and spacecraft industries. As is well known, the U.S. launch market share fell from ∼80% in the late 1970s to less than 20% worldwide in 2002.

In the last three decades, only one new government-sponsored and one new largely commercial-sponsored booster engine have been developed and gone through flight certification. These are the SSME and the Boeing RS-68, respectively. The SSME was, of course,



Robert L. Sackheim just recently retired from NASA, after seven years as the Assistant Director and Chief Engineer for Propulsion at NASA's George C. Marshall Space Flight Center (MSFC). He currently is an independent consultant and an adjunct professor at the University of Alabama, Huntsville (UAH). He holds a B.S. degree from the University of Virginia, an M.S. degree from Columbia University, and has completed all doctoral course work in chemical engineering at the University of California in Los Angeles. He joined MSFC after 35 years in various technical and management positions with the TRW Space and Electronics Group. His awards and honors include the AIAA James Wyld Award for outstanding technical contributions to the field of rocket propulsion, as well as 12 NASA Group Achievement Awards. While at TRW he received three annual Chairmen's Awards and a TRW patent of the year award. He is a fellow of AIAA and was elected in 2000 to the National Academy of Engineering. He also received the AIAA Sustained Service Award in 2000. The Alabama/Mississippi section of the AIAA awarded him the Martin Schilling Award for outstanding service to the section, the Herman Oberth Award "For Outstanding Individual Scientific Achievement in the Fields of Astronautics and Space Sciences," and the Helgar Toftoy Award for outstanding technical management. He recently received an award from the Association of Aeronautics and Astronautics of France for "High Quality Contributions to the Propulsion Field." In 2001 he was awarded the NASA Medal for outstanding technical leadership, and in 2003 he was awarded a presidential citation for meritorious federal executive service. He has served on a number of NASA boards, including the Shuttle Independent Assessment Team (SIAT), the Mars Climate Orbiter Mishap Investigation Board, and the Mars Polar Lander Mishap Board. He has authored more than 250 technical papers. He also holds nine patents for spacecraft and/or launch vehicle propulsion and/or control systems technology.

Received 21 February 2006; revision received 7 September 2006; accepted for publication 5 September 2006.

*Assistant Director and Chief Engineer for Propulsion (retired), Office of the Center Director. AIAA Fellow.

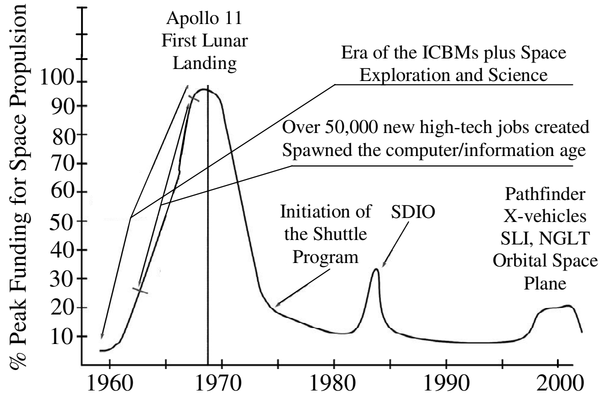


Fig. 1 Historical trends in national rocket propulsion funding as a percentage of the peak funding for Apollo by year.

originally developed for the space shuttle in the 1970s. Some significant upgrades have been incorporated since the original certification for flight, but their modifications have been to increase reliability and safety and to somewhat improve reusable launch-vehicle (RLV) operability [i.e., increase mean time between refurbishment (MTBR) from one mission to another, on the order of 10–15 RLV flights]. Very little advancement in reducing costs or increasing performance has been achieved. In fact, if anything, performance and cost have been sacrificed to increase safety and reliability.

The RS-68 engine was developed by Boeing/Rocketdyne as a low-cost expendable booster engine for the Delta IV evolutionary expendable launch vehicle (EELV). However, from published data by the contractor, it did not meet its original target goals in nonrecurring and recurring (unit) costs and in performance. Engine performance of the RS-68 is comparable to that of the 1960s period Saturn V second- and third-stage J-2 engines, both of which were simple open-cycle, gas-generator powered designs. The only claim for any advancement that can be made by the developer for the RS-68 is the much higher thrust level and the incorporation of modern design and manufacturing techniques. This would be very beneficial in production runs of 30–50 engines per year, but it appears that in the near-term space marketplace only about 8–10 engines per year will be needed to meet current demand. This will produce almost no unit cost advantage over older engines that are available anywhere in the world today. A summary description of this engine is given in Sec. V.

While the United States has developed almost no new rocket technology during the last 30-plus years, the rest of the world has been quite busy doing exactly the opposite. The space faring nations in Europe, Asia (including India), the Middle East, and the former Soviet Union are believed to have developed between 40 and 50 new, high-performance engines of almost every type of combustion/propellant system or power cycle that is conceivably enabled by today's rocket technologies. Based on these observations, it is probably no coincidence that the total U.S. share of the space launch market and U.S.-built launch-vehicle reliability has eroded badly in the last 40 years. In the commercial space marketplace alone, the United States captures only about $1 to 2 billion out of a potential worldwide commercial launch market of $8 to 10 billion per year as of today.

A similar trend has also been observed in the upper-stage and in-space propulsion technology product development areas. Advancements in both will be greatly needed to enable future exploration missions. Most of the U.S. in-space propulsion developments in recent times have been privately funded with some support from the government. Even for these technology investments, most of the government-sponsored projects were stopped for one reason or another before any significant advances in technology readiness could be achieved. Only now has the government begun to realize that the paucity of new technology developments for the high leverage of increased-performance in-space propulsion systems has greatly hampered the growth of Earth-orbital and deep-space missions. Many of these high-performance in-space propulsion technologies are literally required to enable many of the future space-exploration missions planned for classes of human and robotic spacecraft, as well as for many national defense missions.

II. Importance and Key Functions of Space Propulsion

Space propulsion systems perform three basic but highly enabling functions for all U.S. payloads and assets that operate from space:

1) They lift the launch vehicle and its payload from the Earth surface-based launch pad and place the payload into low Earth orbits (LEO).

2) They transfer payloads from LEOs into higher orbits, such as geosynchronous, or into trajectories for planetary encounters (including planetary landers, rovers, and sample return launchers if required).

3) Finally, at the mission operational location, they provide thrust for orbit maintenance, rendezvous and docking, position control, station-keeping, and spacecraft attitude control (i.e., proper pointing and dynamic stability in inertial space).

Each of these sets of space propulsion functions provides unique, mission-enabling capabilities. The only type of rockets capable of providing high thrust forces required for an Earth-to-orbit (ETO) launcher are those that convert thermal energy to kinetic energy. Only two such technical approaches are conceivable today: 1) chemical combustion and 2) nuclear reactor thermal power. Nuclear thermal reactor rockets are unacceptable because they are open-cycle devices that emit radioactive materials. Thus, chemical rockets will launch NASA payloads into LEO for the foreseeable future.

It is also important to recognize that all flight experience with chemical in-space propulsion systems [Apollo, satellites, planetary spacecraft, shuttle reaction control system (RCS), and space station] has been with storable propellants. Other than the two- or three-burn Centaur/RL-10, the United States has never flown an in-space propulsion system that uses cryogenic propellants, even though many of the leading propulsion-system candidates for future exploration missions have cryogenic upper-stage systems designated in their conceptual designs. The long-term storage and transfer of advanced cryogenic chemical propellants present a wide range of technical challenges that must be addressed in the near term. The nation's ability to mature these technologies to a level that assures flight readiness will be one of the key factors in ensuring the successful execution of future space-exploration missions.

Once a spacecraft has reached LEO, high thrust is no longer required for in-space operations. Low-thrust systems operating for a long time can achieve the same result as a short-duration, high-thrust system. Frequently, the mission designer must trade off trip time and propellant mass. However, for missions relating to transfer of crews from Earth to other destinations, long-trip times are not an option. Once a spacecraft returns to a gravity well, like the moon or Mars, high-thrust chemical propulsion is again required (or aerocapture as a possible option if it is developed, proven, and applicable).

For all of the applications inherent in human exploration of the moon and Mars, space propulsion in general and chemical propulsion in particular will be a technology area critical to success. Space propulsion for exploration is used to refer specifically to these functions: 1) safe and reliable access to space; 2) in-space maneuvering: orbit transfer, trajectory insertion, and attitude control (position control, rendezvous, and docking); and 3) ascent and descent propulsion for human operations at the moon and Mars.

These functions are the key capabilities by which the nation's long-term vision for space exploration will be enabled and that must be made available to achieve the required exploration mission success. This paper will clearly and accurately illustrate that these capabilities have been allowed to atrophy in the United States, jeopardizing the potential success of the future planned human exploration program.

The mass involved with human space exploration means that the primary focus of space propulsion must be on both rocket engines with thrust levels greater than 100,000 lbf (45,359 kgf) and in-space/

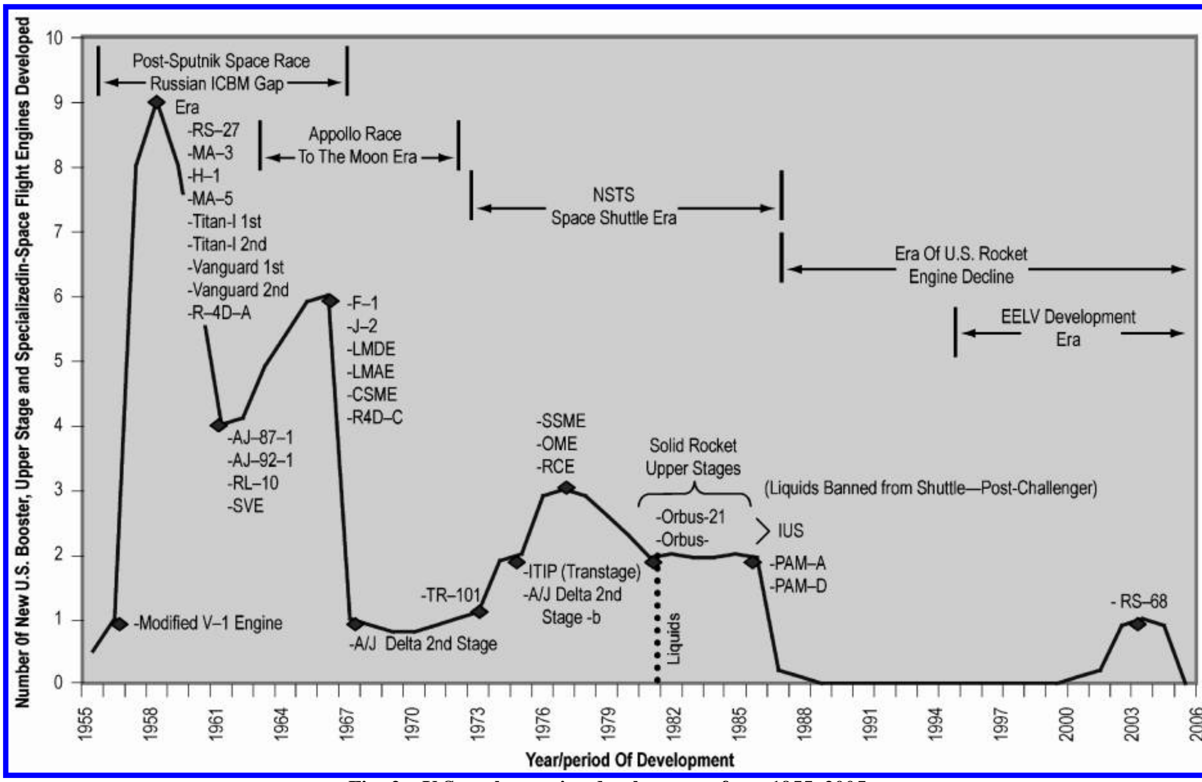


Fig. 2 U.S. rocket engine developments from 1955–2005.

lander propulsion systems with thrust levels of 25,000–50,000 lbf (11,340–22,680 kgf). Figures 2 and 3 indicate that the United States has not developed such rocket engines since the development of the SSME and the RL-10 for in-space/upper-stage vehicles (also refer to Fig. 1). Table 1 summarizes the current capability available today for upper-stage and in-space propulsion applications. Table 1 also reinforces the point that no significant high-thrust upper-stage or in-space propulsion systems have been developed in the last decade. The only new liquid rocket engines in this size range developed in the United States and fielded after 1980 are a modification of a rocket engine developed by Russia (i.e., RD-180) and the RS-68 engine by Rocketdyne.

III. Considerations for the Nation's Space Program

All of the lessons learned from the only four human space flight programs (Mercury, Gemini, Apollo, and Shuttle) in U.S. history demonstrate the value of a strong, knowledgeable, highly experienced, and successful agency and a nationwide propulsion engineering team. The new emerging space-exploration activities need to retain, use, and build on what remains of this previous strong propulsion technical knowledge base.

Based on a detailed review of propulsion-system historical development data, it is apparent that hardware failures will occur

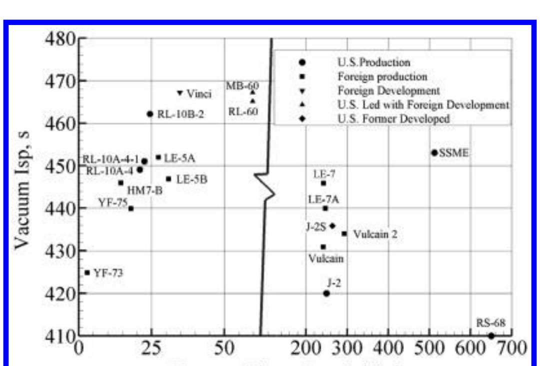


Fig. 3 Existing LO_2/LH_2 engine production level designs.

during the development of any new propulsion system (Table 2). These failures almost always result in the redesign or modification and then repair and replacement of the failed propulsion hardware. As more and more tests are performed, supported by accurate and validated analytical models, design engineers gain a better understanding of the propulsion-system characteristics and are able to reduce the frequency of test failures by applying lessons learned and by building on the knowledge gained from previous successes and failures. This development cycle allows the design team to optimize the system design, while simultaneously gaining invaluable hardware experience that will enable them to reduce the development time of future systems. This cycle of "test–fail–fix–test" also clearly demonstrates that the hardware is the key driver for development and the associated costs. These costs must be minimized for future developments through the judicious, intelligent, and logical use of the advanced computational models and codes and modern high-speed computer capabilities now available. The ideal model for this balanced engineering approach is illustrated in Fig. 4.

This observation is further substantiated by analysis of historical rocket and jet (gas turbine) engine development costs. Close examination of the detailed cost distribution for major engine development programs, compiled by the Marshall Space Flight Center (MSFC) in the early 1990s, reveals a remarkable consistency. This analysis was conducted for the F-1, J-2, SSME, RL-10A-3, Space-Shuttle OMS, and the F100 jet engines. The results clearly show that about 50% of the development cost is the hardware, whereas the remaining portion is evenly split between the labor elements of test, engineering, and management.

The obvious conclusion is that major reductions in development costs and schedules should be achievable if the cost of the development hardware can be reduced (i.e., design the hardware and program for low cost instead of maximum performance). This observation is particularly true if the hardware is both low in cost and relatively mature (i.e., base the new design approach on existing design databases), which was not the case for the Saturn V, space shuttle, or the RL-10A-3 upper-stage engines, so that relatively few hardware failures occur. The reduction in failures will also result in fewer test cycles, further reducing costs and schedule for the engine full-scale development (FSD) program. For propulsion activities that do not have past hardware development experience to build on, a

Table 1 Summary of in-space rocket engines and thrusters currently in place

Type	Manufacturer	Propellants	Nominal thrust range, lbf (kgf)	Nominal I_{sp}, s at vacuum	Nominal operating life	Engine weight, lb (kg)	Status	Date of development complete
Spacecraft apogee or large Delta-V engines (chemical engines/ thrusters)	Aerojet, Northrop-Grumman, IHI, AMPAC, EADS	Storables	50–300 (23–136)					
R4-D class	Aerojet, EADS	MON/MMH	90–100 (41–45)	311–328	2–4 h	8 (4)	Flown	1956
MMBPS class	Northrop-Grumman (Formerly TRW)	MON/MMH	100 (45)	315	2–4 h	7.5 (3)	Flight qualified	1970
DMLAE	Grumman-Grumman (formerly TRW)	MON/N_2H_4	105 (48)	315–335	2–4 h	10 (4.5)	Flown	1996
DMLAE	EADS	MON/N_2H_4	100 (45)	310	2–4 h	10 (4.5)	Flown	1972
DMLAE	AMPAC	MON/N_2H_4	100 (45)	310	2–4 h	10 (4.5)	Flown	1994
DMLAE	IHI	MON/N_2H_4	100 (45)	310	2–4 h	10 (4.5)	Flown	1979
Bipropellant RCS class	Aerojet, AMPAC, EADS	MON/MMH	5–2.2 (2–1)	250–300 depending on duty cycle	1–2 h	2.5 (1)	Flown	1982
SCAT Bimodal thruster	Grumman–Grumman	MON/N_2H_4	14–4 (6–2)	320–310	10 h	4 (2)	Flown	1997
Monopropellant hydrazine (N_2H_4)	Grumman–Grumman, NGST, Aerojet, AMPAC, EADS, IHI	Catalytically decomposed N2H4						
RCS-class thrusters	Grumman–Grumman, NGST, Aerojet, AMPAC, EADS, IHI	Catalytically decomposed N_2H_4	0.1–25 (0.05–0.11)	150–230 depending on duty cycle	60,000 s steady-state 1×10^6 pulses	0.5–10 (0.23–4.5)	Flown	1967
Delta-V class thrusters	Grumman–Grumman, NGST, Aerojet, AMPAC, EADS, IHI	Catalytically decomposed N_2H_4	25–300 (11–136)	225–235	10 h 500 pulses/off pulses	10–50 (4.5–23)	Flown	1970
Planetary lander class	Aerojet	Catalytically decomposed N_2H_4	500–700 (227–317)	240	5 min and throttle 10:1	20 (9)	Flown	1975
OME class	Aerojet	MON/MMH	11,000 (4990)	330	Reusable on shuttle	220 (100)	Flown	1978
RL-10A-1	Pratt & Whitney	O_2/H_2	15,000 (6804)	422	1200 s	300 (136)	Flown	1960
RL-10A-3	Pratt & Whitney	O_2/H_2	15,000 (6804)	427	1200 s	300 (136)	Flown	1965
RL-10A-3-1	Pratt & Whitney	O_2/H_2	15,000 (6804)	431	1200 s	300 (136)	Flown	1967
RL-10A-3-3	Pratt & Whitney	O_2/H_2	15,000 (6804)	442	1200 s	300 (136)	Flown	1969

(continued)

Table 1 Summary of in-space rocket engines and thrusters currently in place (Continued)

Type	Manufacturer	Propellants	Nominal thrust range, lbf (kgf)	Nominal I_{sp}, s at vacuum	Nominal operating life	Engine weight, lb (kg)	Status	Date of development complete
RL-10A-3-3A	Pratt & Whitney	O_2/H_2	16,500 (7484)	444	1200 s	300 (136)	Flown	1972
RL-10A-4	Pratt & Whitney	O_2/H_2	20,800 (9435)	449	1200 s	370 (168)	Flown	1975
RL-10A-5	Pratt & Whitney	O_2/H_2	14,560 (6604)	368	1200 s	316 (143)	Flown	1978
RL-10A-4-1	Pratt & Whitney	O_2/H_2	22,300 (10,115)	451	1200 s		Flown	1982
RL-10B-2	Pratt & Whitney	O_2/H_2	24,750 (11,226)	466.5	1200 s		Flown	1998
Delta II second stage	Aerojet	MON/A-50	10,000 (4536)	315	30 min	220 (100)	Flown	1984
RD-42	Aerojet	MON/MMH	200 (91)	280	1 h	10 (4.5)	Flown	1978
RD-40	Aerojet	MON/MMH	900 (408)	295	Reusable on shuttle RCS (indefinite)	15 (7)	Flown	1977
LMDE	Grumman–Grumman, (NGLT) (formerly TRW)	MON/A-50	10,000–1000 (4536–454)	305–280	2000 s	300 (136)	Flown on all Apollo lunar landing missions	1967
Delta 3910 second-stage engine	NGST	MON/A-50	9800 (4445)	308	2000 s	245 (111)	Flown	1974
RS-4-1	Rocketdyne	MON/MMH	4000 (1814)	305	200 s	30 (14)	Flight qualification Peace Keeper fourth stage axial Delta-V engine	1988
RL-60	Pratt & Whitney	O_2/H_2	60,000 (27,216)	460	1000 s	700 (318)	On hold	Not complete
MB-XX	Rocketdyne, MHI	O_2/H_2	35,000–69,000 (15,876–31,298)	462	1000 s	600–1200 (272–544)	In test	Not complete
Electric thrusters								
Resisto-jet	NGST, Aerojet	N_2H_4 NH_3	0.1 (0.045)	280–300	100 h	4 (1.8)	Flown	1980
ARC-jet	Aerojet	N_2H_4 NH_3, H_2	0.05–1 [1–25 kW]	550–900	100 h	5–25 (2.3–11)	Flown	1986
ION thrusters	NASA, Aerojet	Xe	0.000001 (0.5×10^{-7})	3000 (3 kW) to 9000 (50 kW)	25,000 h	30 (14)	Flown	1988
HALL thrusters	Russia, NASA, Aerojet, Busck	Xe	0.0001 (4.5×10^{-5})	1200 (0.5 kW) to 3500 (20 kW)	100 h	15 (6.8)	Flown	1994
Pulsed plasma	Europe, Russia, Aerojet	Teflon	0.0000001 (0.5×10^{-8})	500	1×10^6 pulses	5 (2.3)	Flown	2000
MPD	NGST	Lithium	10,000 (4536)	5000	1×10^6 pulses	30 (14)	Preliminary development	Not flown
PIT	NGST	Decomposed N_2H_4 NH_3, Argon, Kr, Xe	0.5 at 300 kW	6,000 at 300 kW	1×10^6 pulses	50 (23)	Preliminary development	Not flown

Table 2 Causes of launch-vehicle failures

	Launch-vehicle subsystem failures (1980–1999)							
Country	Propulsion	Avionics	Separation	Electrical	Structural	Other	Unknown	Total
United States	15	4	8	1	1	1		30
CIS/Soviet Union	33	3	2			1	19	58
Europe	7	1						8
China	3	1			2			6
Japan	2					1		3
India	1	1	1	1		1		5

modified Apollo approach of more extensive hardware testing, but guided by today's more advanced modeling capabilities is highly recommended to minimize the potential for flight failures.

The development of advanced software simulation tools has helped to reduce cost and development time by providing the engine design team with the capability to perform dynamic system analysis before hardware fabrication. These tools provide valuable design insight and can lead to rapid system optimization during the early phases of a development program. This optimized system design approach should result in an engine design that is inherently more reliable with a significantly lower development cost. The ability to adequately model hardware and systems also speeds up the whole development activity and generally tends to reduce the number of "test–fail–fix and redesign" cycles.

This observation on the cost-driving characteristics of the hardware applies to the recurring/operational costs for expendable launch vehicles as well. The average recurring cost breakdown for the Atlas, Centaur, Delta II, Titan II, and Titan IV launch vehicles is ~71% for the vehicle hardware. Of that hardware, the next level of breakdown shows that ~54% of those costs are driven by propulsion elements (engines, strap-on boosters, etc.), not including the tanks, which are classified as part of the vehicle structure for bookkeeping purposes. So here again, a concerted effort to reduce engine costs will also greatly reduce expendable launch vehicle recurring and development costs. As a further cost-reduction observation, it can be readily shown that engine costs are mostly a function of operating chamber pressure (p_c) and parts count, which basically is a strong function of the type of power cycle (i.e., pressure-fed, gas generator, expander, or staged combustion).

It should be readily apparent from the above key points that simplified engine designs and existing databases, in combination with effective analysis/modeling tools, should go a long way toward reducing the development and recurring costs for booster and upper-stage engines for future expendable heavy-lift launchers, and in-space and descent/ascent engines and systems for human spacecraft. However, it must also be cautioned that if the U.S. government does

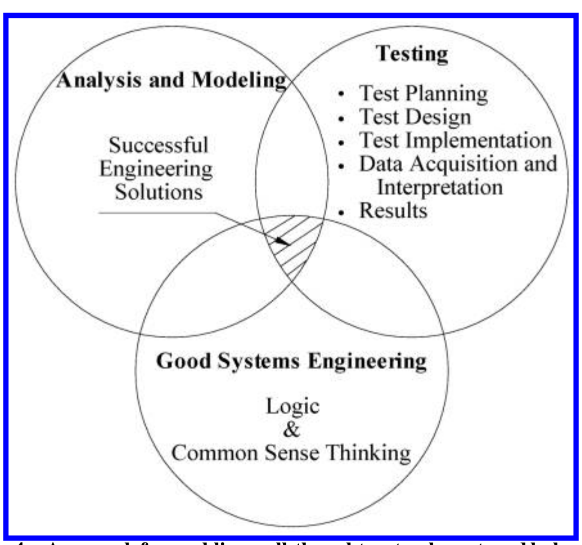


Fig. 4 Approach for enabling well-thought-out, relevant, and balanced engineering solutions.

not act to shore up and sustain the rapidly diminishing, but still above "critical mass," national rocket propulsion capabilities, it will soon be lost to the next generation of engineers that will subsequently be entering into the propulsion community. A good example of this problem was the major unplanned cost overruns (50–100%) experienced on the recent RS-68 booster engine development program.

Returning to the big picture of the state of propulsion technology today, it is important to note that many of the new booster and upper-stage engines that were initiated by NASA and the U.S. Department of Defense (DoD) were stopped or reduced to the point of ineffectiveness because the commitment and funding required to solve various complex problems that appeared, as they always do, early in development and technology projects were lacking. Furthermore, underlying the basic lack of propulsion and space transportation technology development activities in the last several decades is the truly critical need to advance research in all areas of chemical (e.g., new higher-energy/higher-density propellants, combustion devices, advanced turbomachines, etc.), high-power electric and plasma (e.g., Lorentz force accelerators, etc.), and advanced nuclear (e.g., fission, fusion, antimatter, etc.) propulsion for the future exploration mission needs.

Even though a significant amount of money has been spent on launch-vehicle, spacecraft, propulsion-system, and engine development programs since the completion of space shuttle development, most of these programs and engine projects were canceled early in the development cycle. The farthest progress that most chemical engine projects achieved was development testing and only one program (besides RS-68) proceeded past the protoqual stage. However, the expertise needed to support future human rating of rocket engines and propulsion systems comes primarily from direct involvement in development, qualification, and flight activities. Because no chemical engine (except RS-68) reached these stages since the shuttle began flying, it can be argued that the funding spent during this time has not provided the needed experience to support future space exploration.

In fact, as the number of new space transportation and propulsion development programs have greatly diminished over the past two-plus decades, the response of the industrial base has been to consolidate and merge and to greatly shrink their combined capabilities and respective work forces. Some of the more recent and more significant consolidations are summarized in Figs. 5 and 6. These consolidations are already indicating an alarming decline in the previously dominant position of the U.S. aerospace industry globally. Further reductions could seriously inhibit the U.S. capability to compete in our domestic, as well as the international, marketplace.

IV. Summary of Lessons Learned from the Apollo and Space-Shuttle Programs

The Apollo series of missions to the moon stands as one of the single greatest achievements in the history of space exploration. Two great super powers, the United States and the Soviet Union, entered into a race to be the first nation to land humans on the lunar surface and then return them safely to Earth. Both nations experienced serious accidents resulting in crew fatalities during the early stages of their respective moon landing programs. This tragic loss of life

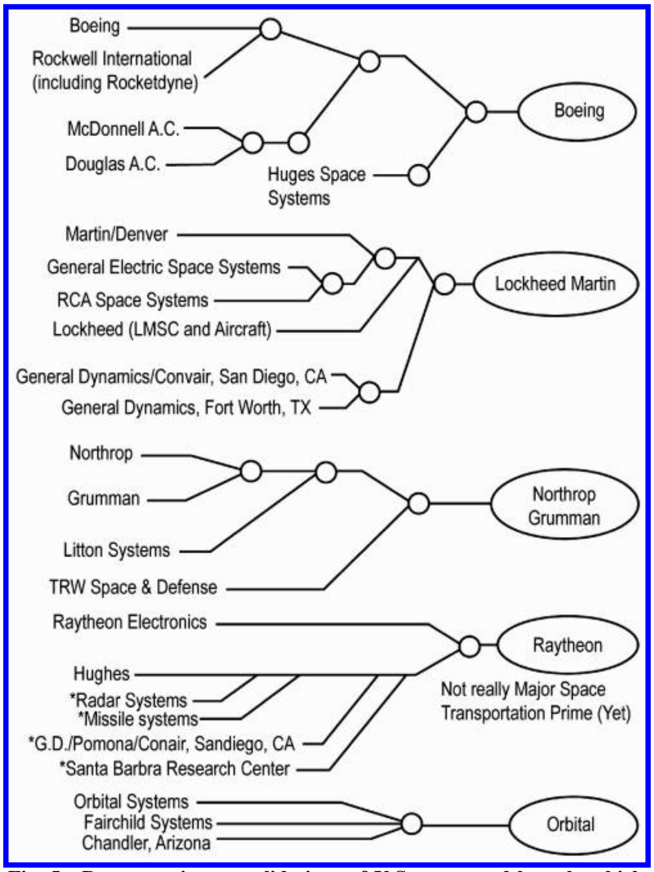


Fig. 5 Recent major consolidations of U.S. space and launch-vehicle contractors.

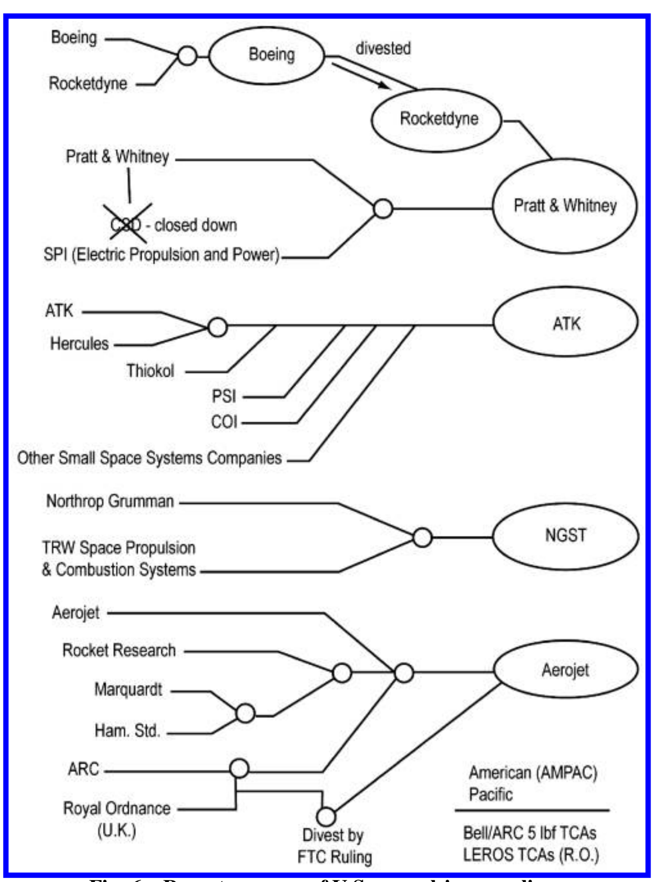


Fig. 6 Recent mergers of U.S. propulsion suppliers.

marred both nations' human lunar landing activities and underscored the high degree of difficulty and risk associated with human space-exploration missions. Because of potential gains in global stature, however, both nations considered winning the high-profile geopolitical cold war race to be well worth the risks and cost, so the race to the moon continued beyond the impacts of the terrible experiences on both sides.

Despite this dedication, the Soviets encountered increasingly difficult technical problems, resulting in significant failures of their moon rocket launcher, the N-vehicle, and additional loss of life. Only the United States was finally successful in landing two men on the moon and safely returning them to Earth, as directed by President John F. Kennedy's national imperative. This unprecedented accomplishment was enabled by an extraordinary program/project management team, supported by an outstanding, highly talented engineering team drawn from all facets of the American aerospace community and backed by the technological and financial resources of the most powerful nation in the world. In the words of former Apollo Program Manager George E. Mueller, "Apollo was the first space system of systems ever created and implemented" [1]. It might now be worthwhile to look back at some of the specific achievements that were required to accomplish this remarkable, singularly successful series of events, in the interest of obtaining some lessons learned that can be applied to future exploration missions. This is especially relevant in light of the recognition that future space exploration by humans will require an extremely complex and interrelated space system-of-systems approach. From launch vehicles through a myriad of in-space vehicles, a complicated array of systems and infrastructure must be designed, developed, and deployed to enable successful completion of all of the missions required to ultimately land humans back on the moon and then on Mars.

As a further basis for using the lessons of the highly successful Apollo program, it must be remembered that there was actually a sequence of 10 successful individual Apollo missions—Apollo 8 through 17. Six of these actually enabled a total of 12 American astronauts to walk on the moon and return safely to Earth. Much was learned from each Apollo mission and applied to the subsequent one, culminating in the final lunar exploration and scientific research mission, the first and only human-based lunar geological survey, performed by the Apollo 17 crew. This is an important precedent for the complex exploration program that will be conducting all of the robotic and human missions planned for over a 30-plus year period and culminating with multiple safe and sustainable landings of humans on the planet Mars for relatively long-duration stays on the surface of the planet.

At the time, Apollo was considered to be the most ambitious engineering project ever undertaken, a task that was considered to be far more complicated than the Panama Canal and the Manhattan Project combined. The idea of designing and building a 36-story-high rocket, loading it with more equivalent energy than an atomic bomb, putting three men on top, launching them to the moon, safely landing, and then returning them to Earth seemed even to some of the excellent engineers of that day to be just about impossible, Mercury-Redstone had just completed its initial suborbital flight in May 1961, and would require nothing short of a miracle to complete. Nonetheless, the task was accomplished multiple times with higher reliability than any rocket in history.

The Apollo spacecraft alone required 500×10^6 man-hours of work. It contained more than 2×10^6 functioning, separate parts that had to survive intense launch and boost vibrations and loads, then function in the weightless void of space, where the temperature range between sunlight and shadow was greater than 600°F (315°C). By far, the most critical technologies that had to be designed, developed, and matured were the rocket propulsion elements required for all of the critical and serial operating modes of the numerous space transportation systems in this unprecedented system of systems. Ten new rocket engines had to be designed, qualified, and human rated from scratch (Table 3); no engine existed anywhere in the world that could operate at the desired design conditions or over the required ranges [e.g., 1.5×10^6-lbf (680,389-kgf) thrust for each of five main

Table 3 Number of rocket engines developed for the Apollo mission

Saturn 5/F-1 engine	F-1
S-2 second-stage S-1VB	J-2
RCS thruster	R4-B
Solid escape tower rocket	ER
SE-7 ullage control S-2	SE-7
SE-8 command module RCS S-14B	SE-8
TRW ullage rocket	UC-1
Service module main engine	AJ10-137
Lunar descent engine	LMDE
Lunar ascent engine	LMAE
Total no. of new engines developed for Apollo	10

booster engines and 10:1 deep throttling for the lunar module descent stage to land on the moon].

This was the all-time peak activity for the then-fledgling U.S. rocket industry, as shown in Figs. 1 and 2. The United States was rapidly developing multiple liquid-stage and in-space propulsion systems required for the Mercury, Gemini, and Apollo programs. One of the primary factors contributing to the successful development of these systems was the knowledge and technical expertise gained during the liquid propulsion engine development efforts required for the Intercontinental Ballistic Missile (ICBM) programs (i.e., Redstone, Jupiter, Juno, Thor, ATLAS, and TITAN I and II). The ability to leverage experience gained during these programs accelerated the development of liquid propulsion-system expertise in the United States. The explicit benefits of this early liquid ICBM engine development experience soon became apparent in the early stages of mission design during the 1960s, because it was soon determined that three different propellant combinations would be required to meet the launch-vehicle gross liftoff weight (GLOW) constraints and the overall performance, reliability, and mission constraints. For the Saturn V first stage, LO_2/kerosene was selected to ensure the highest propellant mass fraction and the minimum structural mass. For the second and third stages, as well as during the translunar injection (TLI) burns, LO_2/LH_2, which had never been flown before, was selected to provide the highest possible chemical performance. This provided the highest possible in-space payload fractions at that time. Spontaneous ignition using hypergolic N_2O_4/50–50 unsymmetrical dimethylhydrazine (UDMH) N_2H_4 yielded the reliability and multiple-start in-space capability required for guaranteed in-space engine start and restart, multiple-pulsing reaction/attitude control, Earth–moon cruise and return for the service module, lunar descent, ascent back to lunar orbit, and return to Earth.

Six different rocket engine companies, two of which are long out of business, and five different prime contractors, most of which are either out of aerospace or completely out of business today, collaborated under contract to NASA to enable the very rapid and highly successful development of the 10 new rocket engines, ranging from 1.5×10^6 lbf (680,389 kgf) of thrust down to $\sim$100 lbf (45 kgf) of thrust (for the RCS thrusters for the Command, Service, and Lunar modules) and the associated vehicle main and auxiliary propulsion systems for Apollo. Approximately 80% of the \$75 billion (in today's dollars) spent for the design, development, and human rating of the entire fleet of Apollo/Saturn systems and hardware was spent on propulsion and associated elements including the test infrastructure to develop and certify systems. Figure 7 shows the dollars spent by NASA for the development of space transportation vehicles since that time for comparison purposes. Also, for comparison purposes, the overall cost of all of the propulsion elements for all of the space-shuttle propulsion elements was about 75% of the complete shuttle program from both a research, development, test, and evaluation (RDTE) and an operational point of view. Examination of historical rocket and jet engine development has shown that roughly 50% of the development cost is hardware related (the rest evenly split between labor, test, engineering, and management). For expendable launch vehicles it was found that $\sim$54% of the hardware recurring costs were driven by propulsion elements and this did not include tanks because they were classified

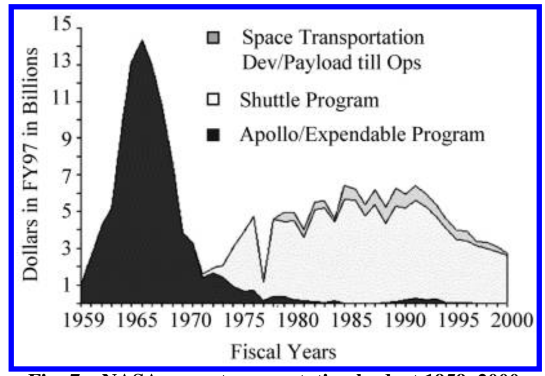


Fig. 7 NASA space transportation budget 1959–2000.

as part of the vehicle structure. Therefore, a significant portion of the cost of vision for space exploration will be the design, development, certification, and manufacture of propulsion hardware and systems.

Of the many incredible development successes leading up to the first landing of humans on the moon, the operational success of all the propulsion elements and the overall demonstrated flight launch vehicle and propulsion-system reliability were truly awesome. As shown in Table 4, every time a Saturn V rocket lifted off for the moon, 86 rockets/thrusters had to work as designed for complete mission success. In many cases, because of weight, volume, and "real estate" access and constraints, many of these rocket engines and/or thrusters could not be redundant (Table 4).

Furthermore, as shown in Table 5, at least 19 serial discrete dynamic events (e.g., staging, rendezvous, landing, ascent, etc.) had to occur flawlessly for complete mission safety and success. It is also remarkable to note that for the 30 or so launches of humans during the Mercury, Gemini, and Apollo programs (all aimed at enabling the United States to land the first humans on the moon) the human-rated launch vehicles (i.e., Redstone, Atlas, Titan II, and Saturn V) demonstrated a flight success record and reliability of 100%, again unprecedented for that period.

It was only through exceptional engineering, scientific, mechanical and electrical skills, innovation, and dedication that this record of success could have been achieved. An awesome government, industry, and academic team was assembled that included some of the greatest technical minds in the country including Werner Von Braun and his team of German rocket scientists from Peenemunde, Joe Shea from MIT, Brainard Holmes, James Webb, T. Keith Glenan, Maxime Faget, Harrison Storms, Joe Gavin, Leland Atwood, and many thousands more who were chosen and trained to "grow up" technically in this highly innovative period of technical achievement.

Many of these people drove themselves beyond normal limits of human endurance to ensure the success of this colossal endeavor, which was a testimony to their dedication to the nation's exploration goal. It is clear that we will need to find such people again to enable the success of the exploration missions planned for the future. This will be difficult due to the decline of the American aerospace base and the diminution of the national technical community's capabilities.

As for the human rating of space launch vehicles and flight systems, some additional experienced-based observations are in order here. It was concluded quickly in the early phases of the Apollo RDTE efforts that more than just analysis or probabilistic/statistical analytical risk assessment would be needed to guarantee the required level of mission success, reliability, and confidence in the design approach or meet the overall safety goals that were required to send humans to the moon and return them safely to Earth. The name of the reliability/mission confidence-building game had to be test, test, test, and then more testing.

In fact, it also became apparent that there are really five basic approaches to "human rating" a system of systems and all of its integral elements. These are as follows:

1) by selecting a sufficiently large number of specimens of the new product design to be tested;

Table 4 Total number of rocket engines/motors per flight of each of Apollo 11–17 missions

S-IC stage	F-1 Engines, $LO_2/RP\text{-}1/1.5 \times 10^6$ lbf—Rocketdyne (R/D)	5
	Retrorockets, solid fuel-Thiokol	8
S-II stage	Ullage rockets, S-IC/S-II interstage 100 lbf—R/D	4
	Retrorockets for two auxiliary propulsion-system modules solid rockets—Thiokol	4
	J-2 engines $LO_2/LH_2/200,000$ lbf—R/D	5
S-IVB stage	Main ullage rockets, jettisonable	2
	Rockets for two auxiliary propulsion systems modules 100 lbf—TRW	8
	Fwd. compartment reaction control engines (pitch) 100 lbf—Marquardt	2
	J-2 engine	1
Command module	Aft compartment reaction control engine (pitch) 100 lbf—Marquardt	10
	Roll engines 100 lbf MARQUARDT	2
Service module	Service module engine (SME) $N_2O_4/A\text{-}50$, 20,000 lbf—Aerojet	1
	RCS engines 100 lbf	16
	Descent engine, $N_2O_4/A\text{-}50$ 10,000–1000 lbf deep throttling—TRW	1
Lunar module	Ascent engine, N_2O_4 3000 lbf Bell/R/D	1
	RCS engines 100 lbf—Marquardt	16
Total		86

2) by conducting a sufficient number of tests on each prototype and flightlike specimen as well as robust subsystem and system-level tests to demonstrate adequate operating and performance capabilities under nominal operational and environmental conditions;

3) by conducting significant off-nominal and/or overstress conditions for testing to demonstrate adequate new product design and system margins;

4) to design the new product systems with as much redundancy as feasible to minimize single-point failures; and finally

5) to use a logical optimized combination of all of the above approaches to establish sufficient confidence in the human-rated capabilities, reliability, and safety of the new products and their associated integrated systems under all anticipated operational conditions.

This was the methodology employed to achieve the confidence in the human-rated capabilities of all the elements and systems associated with the Apollo program and was certainly a major factor in achieving such a stunning series of mission successes and an incomparable record of mission reliability and safety. The human-rating approach used during Apollo, and not strict reliance on two-fault tolerance, was the paramount approach to achieving the reliability and performance for the propulsion and space transportation elements of Apollo.

Because of the need to save weight on flying space vehicles by maximizing propulsion performance, propellant/propulsion hardware mass is typically 80–90% of the takeoff GLOW for any mission, the propulsion engines and all other elements have to operate at maximum performance and at the maximum acceptable limits of their thermal, structural, and dynamic capabilities. The way to resolve these dangerous issues for new rocket propulsion systems is exactly as was implemented for the Apollo program. Find and employ the most knowledgeable, experienced, and brightest propulsion engineers available at the beginning of the new program and in the early design and development phases, and then continue to bring new, very smart, but still inexperienced engineers onboard so they can be trained and eventually provide equally valuable contributions and make good technical decisions before the program enters the flight phases.

An example of the benefits of using the above approach for designing and certifying high-reliability human-rated flight systems can, perhaps, best be illustrated by examining the reliability of the world's launch vehicles. The launch and boost phase of a space flight program is considered to be the most dangerous phase of any space mission, because of the huge amount of propulsion power that must be generated, controlled, and released. The launch vehicle must add $\sim$9100 m/s to take a payload from Cape Canaveral/KSC, Florida to a minimum stable Earth-parking orbit.

With this in mind then, it is very interesting to review the launch vehicle/phase success rate for all space-faring nations for about the last 30 years (1970–1999). There have been about 3450 launches, both manned and unmanned, during this period. The success rate for all these launches combined has remained essentially flat at about 93 to 94%. If one were to look just at the human space launches worldwide during this period, a very different conclusion emerges. To date, the world has launched about 242 human space flight missions with only one launch failure of a reusable launch vehicle (i.e., STS-51L or Space Shuttle Challenger) being reported. This results in a human space launch phase success rate of 99.6%, or very close to the minimum value of 99.8% sought by Von Braun's Saturn V rocket team essentially broken down as follows:

1) about 242 total manned space launches worldwide (U.S., USSR/Russia, China);

2) 241 successful launches (99.6%);

3) STS51L (Challenger) lost;

4) 238 successful landings (98.8%);

5) STS 107 (Columbia) lost on reentry;

6) Soyuz-1 lost on landing (tangled chutes);

7) Soyuz-11 crew lost on reentry (asphyxiated);

8) total manned flight success rate is about 98.3%;

9) worldwide total manned and unmanned launch success rate from 1970 to 1999 (about 3449 launches) essentially unchanged at about 94%;

10) a challenge is to increase manned flight safety given a 30-year history of unmanned launch success rates that are essentially flat.

This comparison of results over the last 30 years of the 20th century (the dawn of the Space Age) shows how much can be achieved by very smart propulsion engineers and managers, combined with a development and certification methodology of stringent repeated testing, and strongly reinforced by a mindset that

Table 5 Number of staging/dynamic events on a nominal Apollo mission

First stage	1
Second stage	1
Third stage	1
Escape tower, jettison	1
CSM to lunar orbit after finishing TLI	1
Interstage separation	1
CSM to LEM docking	1
LEM removal from launch shroud	
LM rotation/reconnect	1
LM separation from CSM in lunar orbit	1
Lunar landing	1
LM egress and surface activities	1
Return to LM ascent stage blastoff	1
Rendezvous with CSM	1
Leave lunar orbit	1
Enter TEI trajectory	1
Stage off service module	1
Earth reentry and kinetic braking	1
Deploy parachutes	1
Total no. of events	19

demands that the development team test as they plan to fly and fly only as they have tested. For example, referring to Table 2, it can be seen that for the last 20 years (1980–1999), 64 out of a reported 114 failures, which equates to about 56%, were caused by propulsion-system failures for all reported launches, worldwide. The next highest cause of vehicle failure was a tie between avionics and separation/staging subsystems at about 10%. Refer to the numbered list above and Table 2 for more detailed information and data on launch vehicles and associated subsystem failure rates and causes.

If one were to examine the development and flight history of the U. S. space shuttle or STS program, very similar lessons about the importance of good propulsion engineering and knowledge, combined with a comprehensive, well-thought-out development and flight certification program would also be apparent for the future missions. The space shuttle was a very different type of human space flight program from Apollo. In some ways it was more difficult due to the reusability requirements and goals (100 flights) established by the program (i.e., it was the first and only reusable launch and astronaut return space transportation system). These requirements were extremely difficult and completely changed the concept of rocket engines, thrusters, and main propulsion systems.

In other ways the space shuttle system could be considered far less challenging than Apollo. The shuttle only had to carry humans to LEO, a distance of 200–400 mi (322–644 km), and no further, whereas Apollo had to travel about 240,000 mi (386,243 km) to the moon, land safely, and return with the human cargo, even though it was an expendable space flight vehicle. In addition, Apollo was extremely weight constrained, much more than the space shuttle human/cargo-carrying STS, which increased the technical challenge.

Here again, the overall cost of the propulsion engines, thrusters, components, and main and auxiliary propulsion systems proved to be much higher than anything else in the system, both for development and flight certification, as well as for flight hardware operation and maintenance.

The overall cost of all of the propulsion elements for all of the space shuttle propulsion elements was about 75% of the complete shuttle program from both an RDTE and an operational point of view. So, as always, the propulsion elements for a major STS, space flight vehicle, or space system of systems are the most difficult to develop, initially the most unreliable, sometimes well into the flight program, and the most expensive hardware throughout all phases of any space system flight program.

V. Current Status of Space Propulsion Technology

In Sec. II we saw the degree to which the success of the Apollo program was tied to successful developments in space propulsion, and in Sec. III, we saw that the nation's current capability in space propulsion is but a shadow of the capability put in place by the ICBM and Apollo programs. A more penetrating look at this capability will illustrate that on which the exploration program can depend today.

At the present time, with the exception of NASA's stated intent to develop a shuttle-derived series of launch vehicles for the exploration mission, there are no plans to replace the Atlas V or the Delta IV launch vehicles until sometime beyond 2020. This is a realistic observation because it is highly unlikely that any technologies currently in development or envisioned for liquid- or solid-propellant all-rocket (noncombined-cycle) propulsion systems for space access would demonstrate the levels of performance improvement or reduction in operational risks and costs necessary to justify the huge cost of an Atlas V/Delta IV or even larger class new-centerline launch-vehicle development. This includes technologies focused on reusable vehicles and propulsion systems.

The very large range of payload weights deliverable to a broad spectrum of orbits made possible by the modularity configurations available for both Atlas V and Delta IV families of vehicles also makes it unnecessary to develop new launch vehicles having slightly higher performance for any given configuration. The yearly costs for these low-intensity, limited flights per year are driven by site maintenance, cadres of trained personnel and managers, realized failures, and acquisition costs of payloads and expendable vehicles. The influence of delivered specific impulse (I_{sp}) on overall mission costs per year is very small, particularly for the first stage.

There are technologies in development that may permit cost-effective upgrades to various system elements of Atlas V and Delta IV, such as upper stages. Most of these technology-based upgrades would be justified based on eliminating critical failure modes or increasing margins against known retained failure modes.

A. Large Launch Vehicles

1. *Delta IV Family*

The Boeing Delta IV family of two-stage launch vehicles uses a common 5-m diameter first stage powered by a single P&W/R/D (Pratt and Whitney Rocketdyne) rocket engine (RS-68) operating on LO_2 and LH_2. The baseline vehicle, designated as Delta IV Medium, has a 4-m diameter second stage powered by a Pratt and Whitney RL-10B-2 engine using LO_2 and LH_2. Three other configurations of Delta IV Medium vehicles offering progressively more payload weight to LEO or geosynchronous transfer orbit (GTO) are in the family. These three vehicles use Alliant Techsystems GEM-60 (60-in diam) graphite-epoxy solid-propellant motors as strap-on boosters.

As shown in Fig. 8, the vehicles are designated as Medium+ (4,2); Medium+ (5,2); and Medium+ (5,4). The first numbers indicate the diameter of the second stage and payload fairing, and the second numbers designate the number of graphite-epoxy motor (GEM) strap-ons. All of the 5-m-diam second stages also use longer propellant tanks. The fifth vehicle in the family is called Delta IV Heavy. The configuration uses three of the common 5-m-diam first stages in parallel. The second stage uses the same 5-m diam, longer tank that is used on the Medium + (5,2) and (5,4) vehicles.

This family of vehicles is stated to have the capability to deliver from 20,000 to 48,000 lb (9072–21,772 kg) to LEO or 9300 lb to 28,000 lb (4218–12,700 kg) to GTO. The propulsion-system elements that essentially control the various configuration's performance and risks are the first- and second-stage engines and the solid-propellant strap-on motors. These are described below.

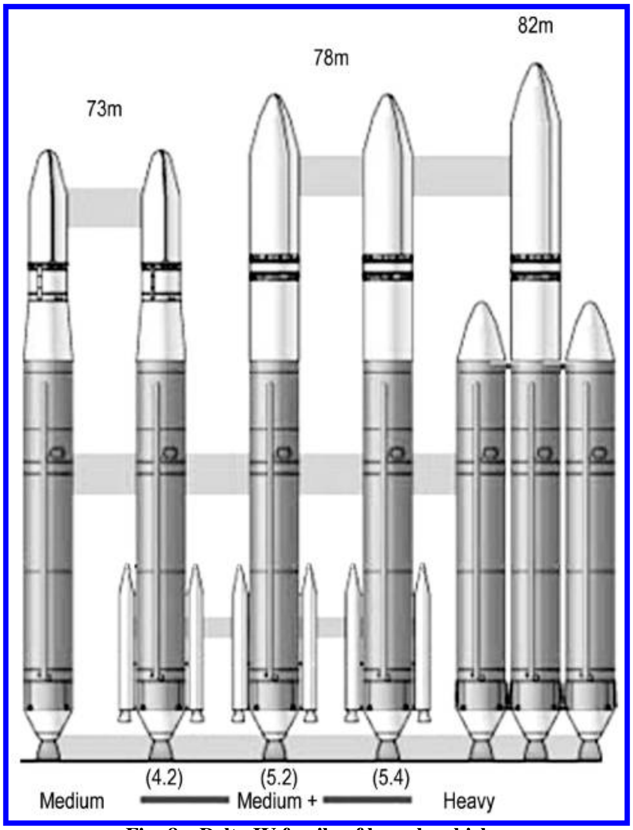


Fig. 8 Delta IV family of launch vehicles.

Table 6 RS-68 engine characteristics

RS-68		
Thrust level	100%	60%
Thrust, vacuum	745 lbf (338 kgf)	440 lbf (200 kgf)
Weight	14,560 lbf (6604 kgf)	——
Thrust, s/l	650 lbf (295 kgf)	345 lbf (156 kgf)
Engine mixture ratio	6	6
I_{sp}, vacuum	410 s	410 s
I_{sp}, sea level	365 s	365 s
Chamber pressure	1410 psia (9722 kPa)	836 psia (5764 kPa)
Expansion ratio (E)	21.5	——

a. Common Booster Core: RS-68 Engine. In the early 1990s, Rocketdyne initiated development of the first new U.S. domestic booster class engine in more than 25 years. This engine, designated as RS-68, uses LO_2/LH_2 propellants and develops 650,000 lbf (294,835 kgf) of sea-level thrust. It is capable of being throttled to 60% of full power level.

The RS-68 engine uses a simple, open gas-generator cycle with a regeneratively cooled main chamber. It was designed, developed, and certified in a little over five years, and flew on the first Delta IV launch in late 2002. The engine has far fewer parts than the SSME, which greatly contributes to its reduced production costs. Only 11 major components are included: the main combustion chamber (MCC), single oxygen and hydrogen turbopumps, gimbal bearing, injector, gas generator, heat exchangers, and fuel exhaust duct. This amounts to an over 80% reduction of parts and a 92% reduction in hand touched labor from SSME. The development cycle time was also much reduced, and the nonrecurring costs were claimed to have been reduced by a factor of 5 over previous cryogenic engines. The engine performance and operating characteristics are summarized in Table 6.

b. Second Stage: RL-10B-2, RL-10 Engine. The RL-10 engine was the world's first LO_2/LH_2 rocket engine operated in space. The engine weighed 289 lb (131 kg) and developed an I_{sp} of 410 s. Since the first successful launch of an Atlas/Centaur RL-10 in 1961, nine additional and different models of the RL-10 engine family have been developed.

The RL-10B-2 is the latest derivative of the RL-10 engine which features a carbon–carbon extendible nozzle. This high-expansion nozzle (285:1) enables the engine to operate nominally with a chamber pressure of 633 psi (4364 kPa) and develops an I_{sp} of 465.5 s. This engine can lift payloads up to 30,000 lb (13,600 kg) and currently powers the upper stage of the medium- and heavy-lift configurations of Boeing's Delta IV launch vehicle in addition to the upper stage of the Delta III. The engine performance and operating characteristics are summarized in Table 7. The full family of flight-certified RL-10XX engines is listed in Table 8 along with their respective key design features.

c. Strap-On Solid Rocket Booster: GEM. The 60-in. (1.5-m) diam GEM motor is a strap-on booster system developed to increase the payload-to-orbit capability of the Delta IV M+ launch vehicles. Two and four strap-on motor configurations of the GEM-60 can be flown on the Delta IV M+ vehicles. The motor features a +5 deg canted, moveable nozzle assembly. This motor is a third-generation GEM with both fixed and vectorable nozzle configurations. Table 9 summarizes the GEM-60 vectorable nozzle operation and performance characteristics.

Table 7 RL-l0B-2 engine characteristics

Characteristics	Details
Thrust	24,750 lbf (11,226 kgf))
Weight	664 lb (301 kg)
Fuel/oxidizer	LO_2/LH_2
Mixture ratio	5.88:1
I_{sp}, vacuum	465.5 s
Starts (total)	15
Service life (total)	3500 s
Expansion ratio	280:1
Length (stowed)\	86.5 in (219.7 cm)
Length (deployed)	163.5 in (415.3 cm)
Diameter (nozzle extension)	84.5 in (214.6 cm)

2. Atlas V Family

The Lockheed Martin Atlas V family of two-stage launch vehicles, shown in Fig. 9, uses a 12.5-ft (3.8-m) diam common-core first stage. The first stage uses a single rocket engine (RD-180) operating on LO_2 and kerosene (RP-1). The baseline vehicle designated as the Atlas V 401 has a 10-ft (3.05-m) diam, 42-ft (12.7-m) long Centaur second stage powered by a RL-10A-4-2 engine. The 400 series has a 13-ft (4-m) diam payload fairing, and the 500 series provides a 16.5-ft (5-m) fairing. The Centaur stage can use either one or two RL-10A-4-2 engines. Depending on the mission, the 500 series can be configured with from zero to five strap-on solid rocket motors (Aerojet, Sacramento). Each motor provides $\sim$254, 000 lbf (1,115,212 kgf) of thrust at liftoff. The 500 series of vehicles has a capability to deliver from 20,000 to 45,000 lb (9072–20,411 kg) to LEO (27.8) or from 8750 to 19,100 lb (3970–8664 kg) to GTO. The propulsion-system elements are described below.

a. Common First Stage: RD-180 Engine. The RD-180 is a two thrust-chamber version of the original Russian RD-170, which is used to power the first stage of the Zenit launch vehicle. The engine features an oxidizer-rich, staged-combustion cycle (ORSC) and burns LO_2 and RP-1 propellants. It has a health monitoring and life prediction system. The two thrust chambers can gimbal ± 8 deg. Minimal interfaces are used between the launch pad and the vehicle.

The engine offers relatively clean operations with oxidizer start and shutdown modes that eliminate coking and the potential for unburned kerosene pollution. Forty to 100% continuous throttling provides the capability for real-time trajectory matching and engine checkout on the pad before launch commit. Key performance and operating characteristics of the engine are summarized in Table 10.

b. Second Stage: RL-10A-4-2 Engine. The RL-10A-4-2 is a LO_2/H_2 closed expander-cycle engine. It is equipped with a single turbine and gearbox that drives the LO_2 and H_2 pumps. Additionally, the engine features a dual direct spark ignition (DDSI) system and can be flown with a fixed or extendible drop-down nozzle skirt. The engine operates nominally with a chamber pressure of 610 psi (4206 kPa) and develops an I_{sp} of 451 s. A summary of the engine performance and operating characteristics is presented in Table 11.

Table 8 RL-10 engine model comparison

Model no.	A-1	A-3	A-3-1	A-3-3	A-3-3A	A-4	A-5	A-4-1	B-2
Vacuum thrust, lbf (kgf)	15,000 (6,804)	15,000 (6804)	15,000 (6804)	15,000 (6804)	16,500 (7484)	20,800 (9435)	14,560 (6604)	22,300 (10,115)	24,750 (11,226)
Chamber pressure, psia, kPa	300 (2068)	300 (2068)	300 (2068)	395 (2723)	475 (3275)	578 (3985)	485 (3344)	610 (4206)	644 (4440)
Thrust/weight	50	50	50	50	54	67		61	
Expansion ratio	40:1	40:1	40:1	57:1	61:1	84:1	4.3:1	84:1	285:1
I_{sp}, s	422	427	431	442	444	449	368	451	466.5
Flight certification date	November 1961	June 1962	September 1964	October 1966	November 1981	December 1990	August 1992	February 1994	May 1998

Table 9 GEM-60 motor with vectorable nozzle

Motor dimensions		Weights, lbm (kgm)	
Motor diameter, in. (cm)	60 (152)	Total loaded	74,158 (33,638)
Motor length, in. (cm)	518 (1316)	Propellant	65,471 (29,697)
Motor performance [73 °F (22.8 °C) nominal]		Case	3578 (1623)
Burn time, s	90.8	Nozzle	2187 (988)
Average chamber pressure, psia (kPa)	818 (5640)	Other	2922 (1326)
Total impulse, lbf-s (kgf-s)	17.95×10^6 (8.16×10^6)	Burnout	8346 (3786)
Burn time average thrust, lbf	197,539	Temperature limits	
Nozzle		Operation	30–100 °F (−1.1–37.8°C)
Housing material	4340 steel	Propellant designation	
Exit diameter, in. (cm)	43.12 (109.52)	QEY, 87% solids HTPB	
Expansion ratio, average	11	Production status	
		Production	

c. Strap-On Solid Rocket Booster. The solid rocket strap-on booster motors have been developed, flight qualified, and produced by Aerojet, Sacramento. The Atlas V family of launch vehicles uses from one to five strap-on solid rocket motors depending upon the mission and launch trajectory requirements. The solid rocket motors are ignited at liftoff and burn for over 90 s, each providing a thrust in excess of 250,000 lbf (113,398 kgf). The motor key design features are summarized in Table 12.

B. Small- and Medium-Sized Launch Vehicles

1. Pegasus

The Pegasus is an air-launched (via a modified Lockheed L-101 I aircraft), three-stage, all solid-propellant, three-axis-stabilized vehicle. It is manufactured by the Orbital Sciences Corporation. The Pegasus-XL vehicle, a “stretched” version of the original Pegasus vehicle, can place a 400–1000-lb (181–454-kg) payload into LEO. The original, or standard, version of the Pegasus was retired in 2000, and only the Pegasus-XL is used today. During a typical flight, the launch aircraft climbs to an altitude of 38,000 ft (11,582 m) and the Pegasus-XL is released from the belly of the L-101 1. The Pegasus-XL begins an unpowered descent at a rate of approximately 60 fps (18 m/s) while the first stage arms and prepares for ignition. The forward velocity of Pegasus during the descent is the same as the launch aircraft, or Mach 0.8 [~524 mph (843 kph)]. After 5 s in free fall, stage one’s solid rocket motor, manufactured by ATK, fires and burns for ~71 s. The Pegasus 22-ft (7-m), delta-shaped wing begins to produce lift as the Pegasus accelerates, and the launch vehicle begins a 2.5-g pull-up. As Pegasus climbs, the booster experiences a maximum dynamic pressure (max-q) of ~1200 lb/ft^2 psi (8274 kPa) approximately 30 s after first-stage ignition. [For comparison, on a typical space shuttle launch, max-q is equal to approximately 600–700 lb/ft^2 psi (4137–4826 kPa).]

The second-stage Alliant Techsystems (ATK) solid-fuel motor ignites ~95 s into the flight at an altitude of 37 mile (60 km), and at ~2 min, the payload fairing is ejected. The second stage flies to an altitude of ~129 mile (208 km) with a velocity of >12,000 mph (19,312 kph). At the appropriate altitude to achieve the designated orbit, the third stage ATK motor ignites and burns for 1 min 6 s to place its payload into orbit.

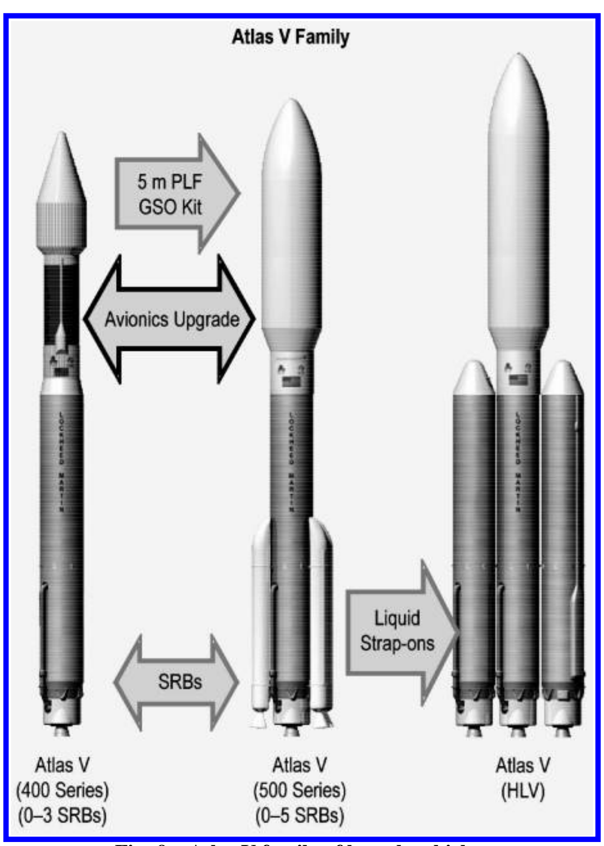


Fig. 9 Atlas V family of launch vehicles.

2. Athena

The Athena program began in 1993 under the sponsorship of LMSC, Sunnyvale, California. The first operational mission of the Athena, Athena I, successfully launched the NASA Lewis satellite of 1750 lb (744 kg) mass into orbit on 22 August 1997. The first Athena II was successfully launched on 6 January 1998, sending NASA’s Lunar Prospector spacecraft of 4350 lb (1896 kg) mass on its mission to study the moon. The Athena propulsion specifics are given in Table 13, although it is believed that this vehicle is no longer available or in production.

Table 10 RD-180 engine characteristics

Nominal thrust	Sea level	860,200 lbf (390,178 kgf)
	Vacuum	933,400 lbf (423,383 kgf)
I_{sp} (sea level)		311.3 s
Vacuum I_{sp}		337.8 s
Chamber pressure		3722 psia (25,662 kPa)
Nozzle area ratio		36.87:1
Mixture ratio		2.72
Length		140 in. (356 cm)
Diameter		118 in. (300 cm)
Throttle range		40–100 %
Total system dry weight		11,889 lb (5393 kg)

Table 11 RL-10A-4-2 engine characteristics

Weight with nozzle	386 lb (175 kg)
Length (approximately)	91.5 in. (232 cm)
Nozzle extension	20 in. (51 cm)
Nozzle area ratio	84:1
I_{sp} vacuum, s	451
Thrust	22,300 lbf (10,115 kgf)
Propellants	LO_2/LH_2
Nominal mixture ratio	5.5:1 O/F

Table 12 Atlas V strap-on rocket motor design features

Designed for maximum payload flexibility	1–5 interchangeable solid rocket motors
SRM size optimized for vehicle design	62 in. (1.57 m) diam, 64 ft (19.5 m) total length
Designed for reliability/robustness	Fixed nozzles
	Monolithic—no. segment joints
	High structural and thermal margins
Designed for producibility	Heritage processes and materials
	Lean production processes, tools, and facilities
	Designed concurrently with airborne hardware
Designed for operability	Fully integrated at the manufacturing site and shipped ready to fly

Table 13 Propulsion specifics of Athena launch vehicle

		CASTOR 120	ORBUS 21D
Stages	Athena I	First	Second
	Athena II	First and second, composite case	Third, composite case
Thrust vector control		Blowdown cold gas-powered hydraulic	Electromechanical
Nozzle		Carbon phenolic	Carbon phenolic
Length, in. (m)		347 (9)	124 (3)
Diameter, in. (m)		92 (2)	92 (2)
Thrust, lb (kg)		435,000 (197,313)	43,723 (19,832)
Propellant		Class 1.3 AP/HTPB	Class 1.3 AP/HTPB
Contractor		Thiokol	Pratt & Whitney

3. *Taurus*

Taurus is an extended-capability, ground-launched version of the Pegasus launch vehicle. It uses three stages of the Pegasus boosted by a large Castor (120) solid-propellant motor. It is designed to launch small and Med-Lite satellites up to 2976 lb (1350 kg) into LEO. Liftoff weight varies between 150,000 and 220,000 lb (68,039 and 99,790 kg). Four variants of the Taurus launch vehicle exist. The smallest version, known as the ARPA Taurus, uses a Peace Keeper first stage instead of a Castor 120 motor. A second size uses the C 120 first stage and a slightly larger Orion 50S-G second stage. The Taurus XL uses the Pegasus-XL rocket motors (Orion 50S-XL and Orion 50-XL) and is considered to be an improved launch-vehicle version. The largest Taurus variant, the Taurus XLS, is a study-phase vehicle that adds two Castor IVB solid rocket boosters to the Taurus XL to improve payload lift capability by 40% over the standard Taurus. For all Taurus configurations, satellite delivery to a GTO orbit can be achieved with the addition of a Star 37 FM perigee kick motor.

4. *Minotaur*

Minotaur is a low-cost, four-stage launch vehicle using a combination of Minuteman II motors and other proven space launch technologies. The Minuteman rocket motors serve as the vehicle's first and second stages, efficiently reusing motors that have been decommissioned for military applications, as a result of arms reduction treaties. Minotaur's third and fourth stages, structures, and payload fairing are common with the Pegasus-XL rocket. Its capabilities have been enhanced by OSC, with the addition of improved avionics systems.

The primary program goal is to provide low-cost, reliable space launch capability in support of U.S. government small-satellite launch requirements. The program was structured to retire most of the development risk with the first mission launch followed by routine launch services operations. The Minotaur is considered a small launch vehicle. It can lift 750 lb (340 kg) to a 400-nmi (741-km), sun-synchronous orbit, which is ~1.5 times the Pegasus-XL capability. The Minotaur can operate with either of two fairings, allowing for the launch of oversized payloads when required. The standard configuration uses a slightly modified Pegasus fairing. It has a dynamic payload volume of about 46 in. (1.2 m) (diameter) by 88 in. (2.3 m) (length). The larger fairing has a dynamic payload volume of about 61 in. (1.5 m) (diameter) by 133 in. (3.4 m) (length).

C. Integrated High-Payoff Rocket Propulsion Technology Activities

The integrated high-payoff rocket propulsion technology (IHPRPT) program was initiated under the Air Force Research Laboratory (AFRL) sponsorship in 1994. It is a joint government and industry effort focused on affordable technologies for revolutionary, reusable, and rapid-response military global reach capability. It addresses sustainable strategic missiles, long life or increased maneuverability, spacecraft capability, launch-vehicle propulsion, and high-performance tactical missile capability. IHPRPT attempts to emulate the IHPTET aircraft engine program, which has been very successful in the development and testing of new turbine jet engine technologies.

Although funding for the program has been limited, contractors have had considerable freedom to develop new technologies with the potential to improve the performance and life of both solid and liquid rocket engines as well as for in-space propulsion technologies. The major difficulty voiced by several of the contractors is that there is no clear definition of near-term and future USAF propulsion needs. Specific performance goals have been established for each element of the program, some of which are summarized in Table 14.

The "building block" approach to planning for technology insertion seems to be working well and furthermore there appears to be excellent cooperation between the USAF and contractors and also between the USAF and NASA. It provides a basis for government and industry to walk in and step on an evolving but common path. Under the IHPRPT program, a new high-performance 40,000-lbf (18,144-kgf) rated LH_2 turbopump, sponsored by AFRL, is currently under study and may be built and tested to validate the improved suite of design/analysis software. Originally, there was also to be an advanced regen-cooled thrust chamber assembly built for a similar purpose, but this was struck from the program due to funding limitations. The only detail design work on the program is directed to the LH_2 turbopump assembly (TPA), and the only testing on the program will be on the LH_2 TPA.

VI. Future Small Launcher Program: FALCON

The FALCON program is funded by the Defense Advanced Research Projects Agency (DARPA), with USAF and NASA support. After the initial 6-month phase I study, four contractors were selected for phase IIb. These were 1) Air Launch LLC; 2) Lockheed Martin Corp.; 3) Microcosm Inc.; and 4) Space Exploration Technologies. Phase II has been divided into three parts, phase IIa covers from phase II authority to proceed (ATP) to preliminary

Table 14 IHPRPT phase I, II, and III goals (2000–2010)

IHPRPT goals	2000	2005	2010
Boost and orbit transfer propulsion			
Reduce stage failure rate	25%	50%	75%
Improve mass fraction (solids)	15%	25%	35%
Improve I_{sp}, s (liquids)	14	21	26
Improve I_{sp}, s (solids)	5	10	20
Reduce hardware costs	15%	25%	35%
Reduce support costs	15%	25%	35%
Improve thrust to weight (liquids)	30%	60%	100%
Mean time between removal (mission life reusable)	20	40	100
Spacecraft propulsion			
Improve $I_{tot}/\mathrm{mass}_{(wet)}$ (electrostatic/electromagnetic)	20%/200%	35%/500%	75%/1250%
Improve I_{sp} (bipropellant/solar thermal)	5%/10%	10%/15%	20%/20%
Improve density I_{sp} (monopropellant)	30%	50%	70%
Improve mass fraction (solar thermal)	15%	25%	35%
Tactical propulsion			
Improve delivered energy	3%	7%	15%
Improve mass fraction (without TVC/throttling)	2%	5%	10%
Improve mass fraction (with TVC/throttling)	10%	20%	30%

design review (PDR). Phase IIb covers from PDR to critical design review (CDR), and phase IIc goes from CDR to flight demonstration of the Smallsat Spacelift mission [2].

The goals for phase IIa were to develop the vehicle design to a PDR level. Concept of operations (CONOPS) was also to be detailed in this phase. Technical risk-reduction activities were defined and executed. The teams interfaced with the launch ranges and then developed refined cost models for the operational system (OS). Independent cost specialists evaluated the competing companies' cost models. The contractors were required to convince the government that they have a high probability of success to continue through CDR and then on to a successful demonstration flight [2]. Figure 10 shows a concept summary description of each of the four phase IIa concepts.

The goals for phase IIb are to develop and evolve the vehicle designs through CDR. During phase IIc, final flight hardware would then be built up and flown no later than fiscal year (FY) 2008. Demonstration flights are expected to carry prototype autonomous flight safety systems (AFSS) and low-cost tracking and data relay satellite system (TDRSS) transceivers (LCT2). After the FALCON program is completed, DARPA hands over the demonstration vehicle system aspects to the Air Force Space Command for operational system development and implementation. It is possible that the vehicle contractor(s) can contract directly with NASA or private entities (e.g., academia, amateurs, and other government agencies) for commercial launches [2].

FALCON is the first concrete program towards the realization of affordable and responsive space lift. Each of the four phase IIa

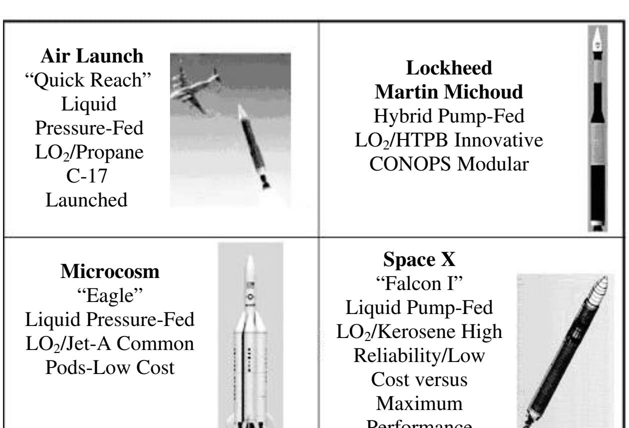


Fig. 10 Vehicle characteristics for the four contractors in phase IIa for the DARPA FALCON program [2].

contractors developed several technologies to meet these goals. Some of the key technologies developed or optimized that could be modified for or transferred to other programs are ablative thrust chambers, self-pressurization systems (VAPAK, Tridyne, etc.), low-cost avionics, hybrid combustion (using a patented staged-combustion concept), and composite tanks. Ablative thrust chambers were used by at least two contractors as an alternative to actively cooled chambers. Composite tanks were also looked at by at least two contractors to reduce weight. Hybrid combustion was successfully demonstrated using a patented staged-combustion concept to achieve both combustion stability and performance comparable to those of liquid-fuel rockets. Common to all contractors was the objective of low-cost operations, which drove each one of them to find innovative ways of streamlining their manufacturing, integration, transporting, storing, and CONOPS processes.

Another observation is that, to different degrees, all of the vehicles studied were modular and easily scalable. These attributes potentially make some of these subsystems great candidates for replacement strap-ons for the Atlas or Delta vehicle families. Also, scaled versions of some of the engines could potentially be used as upper-stage engines. FALCON promises to be the beginning of a new approach to obtain low-cost versatile space launch vehicles from inception of the design. At the time this paper was completed, only two FALCON phase IIa contractors, Air Launch and Space X, have been selected by DARPA to go forward to the phase IIb flight demonstration.

VII. Potential New Space Propulsion Systems

A. Boost Engines

Four competing booster engine design and development programs were initiated and funded under the Space Launch Initiative (SLI) Program at the Marshall Space Flight Center in 2001. Boeing Rocketdyne (now Pratt and Whitney Rocketdyne) had two of the designs, designated as the RS-83 (660,000-lbf (299,371-kgf) thrust LO_2/LH_2) and the RS-84 [1.05×10^6-lbf (476,272-kgf) thrust LO_2/RP-1]. TRW Space and Electronics (now Northrop and Grumman) offered a LO_2/kerosene-fueled engine in the 1×10^6-lb (453,592-kgf) thrust class, which they named the TR-107. The cooptimized for reusable applications (COBRA) was a LO_2/hydrogen-fueled engine in the 600,000-lb (272,155-kgf) thrust class that was to be designed and developed by a joint venture between Pratt and Whitney and Aerojet.

1. *RS-83 LO_2/LH_2 Engine*

The RS-83 was a staged combustion LO_2/LH_2 booster engine system with a single fuel-rich preburner in place of the dual individual preburners used on the SSME. Taking lessons learned

Table 15 RS-83 engine characteristics

Thrust, sea level	663,800 lbf (301,000 kgf)
Thrust, vacuum	749,600 lbf (340,000 kgf)
I_{sp}	445.7 s vacuum
Chamber pressure	2800 psia (19,305 kPa)
Mixture ratio	6.9 S.L/6 Alt
Engine T/W	55 S.L.
Area ratio	40:1
MTBOH/life	50/100 missions
Catastrophic reliability	0.999958
Throttling	50–100% thrust

from the first-generation reusable launch engine, SSME, which is also manufactured by Rocketdyne, the RS-83 engine was designed to be simpler to build and maintain and to have improved controllability and increased reliability. Advanced design features included turbopumps with easy access and fabrication techniques using selectively net-shaped components made through the technology of powder metallurgy. The engine design, performance, and operating characteristics are summarized in Table 15.

The program plan focused on the early development of critical engine components with the overall goal of identifying and reducing the risk associated with the development and testing of these elements. The engine design team identified five critical component risk-reduction tasks: 1) hydrogen compatible materials, 2) turbine damping, 3) subscale liquid preburner, 4) electromechanical actuator (EMA) sector ball valve, and 5) integrated vehicle health monitoring (IVHM) safety/prognostic algorithms.

Unfortunately the RS-83 design and development program was canceled by NASA along with the other three large reusable booster engine programs, when NASA redirected new program goals toward the vision for space exploration in 2004.

2. *COBRA LO_2/LH_2 Engine*

The goal of the COBRA engine program was to produce a rocket engine prototype that would be simple to operate, provide high reliability with long life and reduce cost through multiple flight operations. COBRA planned to incorporate a reusable, hydrogen-fueled liquid booster engine with a thrust level of 600,000 lbf (272,155 kgf). The COBRA engine design consisted of a single fuel-rich preburner, staged-combustion engine using LO_2 and LH_2 as propellants. The engine was to be designed to provide a 100-mission life span with a 50-mission maintenance checkup interval. The proposed engine key design features are given in Table 16. Unfortunately this design concept also became a victim of NASA program redirection (i.e., 2nd SLI/NGLT cancellation).

3. *RS-84 LO_2/RP-1 Engine*

The RS-84 engine program was proposed as the first U.S. reusable hydrocarbon-fueled, oxygen-rich staged-combustion liquid rocket engine. One of the primary goals of the engine development effort was to develop a highly reliable and low-cost maintenance engine for the next-generation reusable launch system. The use of kerosene results in a smaller fuel-tank volume to permit greater propulsive force than other technologies. That benefit translates to more compact engine systems, easier fuel handling and loading on the ground, and shorter turnaround time between launches. All these gains, in turn, reduce the overall cost of launch operations, making routine space flight cheaper and more attractive to commercial enterprises. In addition, because it is not a cryogenic (or extremely cold) fuel like hydrogen, the propulsion system does not require insulation for propulsion-related ducts, valves, lines, and actuators—saving weight and cost. Table 17 shows the proposed attributes of the RS-84 engine development effort. Again, this program was canceled along with the others.

4. *TR-107 LO_2/RP-1 Engine*

A primary goal for NASA's SLI contract for the TR-107 program was to continue development of an engine that could increase the

Table 16 COBRA engine characteristics

Characteristic	Value
Propellant type	H_2O_2
Mixture ratio	6
Vacuum thrust	600,000 lb (272,000 kgf)
Sea-level thrust	492,590 lb (223,435 kgf)
Vacuum I_{sp}	454.7 s
Sea level I_{sp}	373.3 s
Chamber pressure	3000 psia (20,684 kPa)
Thrust/weight (vac)	75
Engine length	180 in. (457 cm)

Table 17 RS-84 engine characteristics

Propellants	LO_2/RP-1
Thrust, sea level	1.64×10^6 lbf (483,000 kgf)
Thrust, vacuum	1.13×10^6 lbf (513,000 kgf)
I_{sp}	324 s vacuum
Chamber pressure	2800 psia (19,305 kPa)
Mixture ratio	2.7
Area ratio	20
Life	100 missions
Throttling	65–100% thrust

safety, reliability, and affordability of next-generation reusable space launch and transportation vehicles. The contract specified that a high-pressure ORSC cycle was to be used with LO_2 and RP. The engine thrust would have been 1×10^6 lbf (453,592 kgf).

In its earliest concept phase, the TR-107 had a central pintle injector for both the MCC and the LO_2-rich preburner. However, performance and risk analyses soon indicated that the main injector should evolve to a distributed coaxial multielement design, given that a single ORSC preburner would be used to drive both the fuel and the oxidizer turbopumps. The oxygen-rich preburner (basically a gas generator) retained the pintle injector because its size was within the pintle injector LO_2-RP test database, and because Northrop Grumman Corporation (NGC) wanted to retain the inherent stability characteristics of the pintle injector. They also wanted the flexibility to make the preburner throttling for mission flexibility and future growth capability.

The TR-107 engine was one of several SLI candidate engines (some of which were described previously) that could be used to provide primary propulsion for the ETO stage of future reusable launch vehicles. Five of the primary technology objectives of this program were accomplished before the SLI efforts were terminated: 1) successful demonstration of a duct-cooled chamber, which eliminates the need for conventional cooling channels; 2) successful demonstration of the preburner pintle injector and propellant mixing, which improves stable throttling and enables slow controlled startup transient performance; 3) establishment of material properties for oxygen-rich compatible materials, which eliminates the need for additional coatings and liners; 4) incorporation of mature combustion devices that minimize parts count for greater reliability and operability; and 5) systems engineered design optimization to minimize cycle pressures, which provides a margin to increase engine life.

The central pintle injector technology is also used in the engines for the FALCON program by Space X (LO_2/RP) and by Air Launch (LO_2/propane). It has been used in every NGC (TRW) in-space bipropellant maneuvering and attitude control (AC) thruster for dozens of major flight satellite programs. Because it can operate with nearly any propellant combination, engines incorporating it are excellent candidates for either the reusable booster or second stage of affordable responsive spacelift (ARES) or future ORS using gas-generator-driven propellant line pumps. In the case of a reusable lifting-body fly-back first-stage vehicle, there could be advantages for a set of throttlable motors along the rear closure of the lifting body. Redundant low-risk throttling gas-generator-driven line turbopumps would be located within the aft region of the lifting body.

5. *XRS-2200 LO_2/LH_2 Engine*

The X-33 program, which began in July 1996, was a 1/2-scale prototype of Lockheed Martin's proposed single-stage-to-orbit (SSTO) concept named the VentureStar. The program was set up as a unique cooperative agreement with Rocketdyne as the supplier of the XRS-2200 linear aerospike engine. Two of these engines were to be used to power the X-33 on suborbital flights to demonstrate the technology needed to proceed with the full-scale VentureStar.

The X-33's aerospike engine, as shown in Fig. 11, was a LO_2/LH_2 gas-generator cycle engine. Each engine had a single oxidizer turbopump, fuel turbopump, gas generator, combustion wave ignition (CWI) system, two aerospike nozzle ramps, 10 thrust chambers (thrusters) per ramp, two redundant engine controller digital interface units (ECDIU), and associated plumbing, valves, and EMAs. The X-33 required two engines in order to provide the needed thrust vector control (TVC) authority. Table 18 summarizes the XRS-2200 key engine characteristics. A picture of the XRS-2200 engine being fired at the Stennis Space Center test stand A-1 is shown in Fig. 12.

The development philosophy of the X-33 program was to accept increased risk to achieve lower costs and a quicker schedule. To do so, the XRS-2200 program relied heavily on the experience gained from Rocketdyne's testing of a linear test-bed engine from 1970 to 1972. Where possible, they used existing hardware and designs. The turbopumps and gas generator were based on J-2 and J-2S engines. Component testing was used for design development, proving margins, and for qualifications. Software was tested with hardware in the loop. Single-engine testing on Stennis Space Center's (SSCs) A-1 test stand was used to verify the design. The two flight engines, in their dual-engine configuration, had a short ignition test and was about to begin acceptance testing when NASA decided not to renew its involvement in the cooperative agreement, because of cancellation of the X-33/VentureStar (Lockheed cancellation) programs.

6. *Fastrac and MC-1 Engines*

In the late 1990s, an in-house rocket engine design and development project was initiated at NASA/MSFC to give the younger newly hired propulsion engineers some real hands-on hardware and design experience. The objectives of this project were to design, build, test, and evaluate a 60,000 lbf $LO_2/RP\text{-}1$ low-cost

Table 18 XRS-2200 engine characteristics

Sea-level thrust	206,800 lbf (93,803 kgf)
I_{sp}	332 s at 100% & 5.5 MR
Mixture ratio	4.5–6.0
Chamber pressure	830 psia (5723 kPa)
Throttling	57–100 %
Differential throttling	±15%
Dimensions forward end	134-in. wide × 90-in. long (340 cm × 229 cm)
Aft end	42-in. wide × 90-in. long (107 cm × 229 cm)
Forward to aft	90 in. (229 cm)

"full-up" rocket engine (prototype level) to be demonstrated in a test-bed facility. Senior MSFC management believed that there were a number of bright and capable young rocket propulsion engineers that had joined the MSFC team but, in a decade, had never participated in, or even seen a real rocket engine hardware development effort. It was felt that a test-bed/prototype engine design and hardware project, conducted entirely inhouse, would be very effective in training and preparing these bright young engineers (under effective guidance of a few MSFC experienced/senior rocket development engineers) to be smart customers for future NASA rocket development projects and flight hardware procurement activities.

The project was named the Fastrac Engine and conducted according to all the NASA design and development ground rules and project criteria. However, just before the formal Fastrac PDR, the engine requirements were changed in a major way so that it could be a candidate for use on the reusable X-34 Pathfinder/X-Vehicle demonstrator program. The name of the engine was then changed from Fastrac to the MC-1. Eventually, the X-34 flight demonstrator program was canceled by NASA, and all further work of any kind on the Fastrac/MC-1 engine was terminated.

However, some very good technology was developed on this Fastrac engine project, the most notable of which was the low-cost Barber Nichols fuel and oxidizer pump assembly. These pumps were well along in their development and readiness for incorporation in the flight MC-1 engine. When the inhouse MSFC project was terminated, the turbopump assembly elements eventually evolved into being used for two small, low-cost launch-vehicle development projects then being jointly sponsored by DARPA, the USAF (SMC), and NASA. Those two projects were intended for use on the FALCON small launch vehicle program, described in Sec. VI.

Barber Nichols TPA elements are currently being used by the Space X version of FALCON to pump propellants to their 75,000 lbf

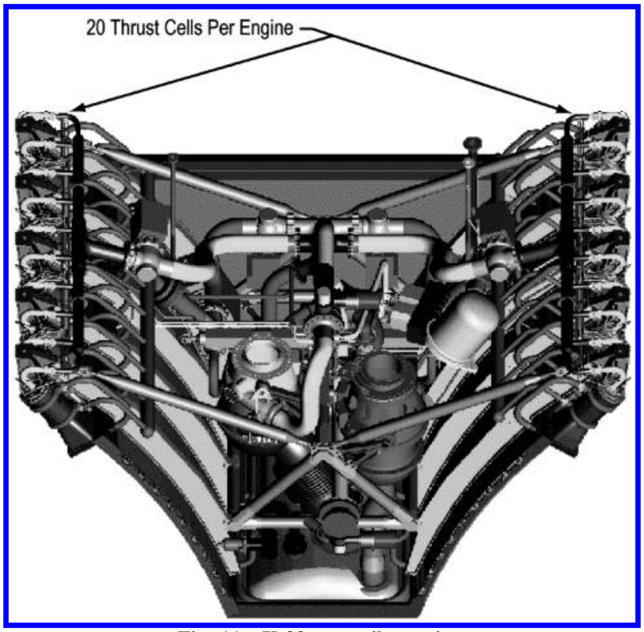


Fig. 11 X-33 aerospike engine.

Fig. 12 XRS-2200 aerospike engine.

(34,019 kgf) LO_2/RP-1 Merlin first-stage liquid booster engine [operating at 850 psia (5860 kPa)], and would be used by Lockheed Martin Michoud to pump liquid oxygen to the first and second stages of their hybrid rocket motor for their version of a low-cost small launch vehicle.

7. IPD: Integrated Powerhead Demonstrator

The IPD engine is a liquid oxygen, liquid hydrogen-fueled full flow staged-combustion reusable technology demonstrator engine. The engine development program was begun by the Air Force in 1994 and more recently has been jointly funded by the Air Force Research Laboratory and the NASA Marshall Space Flight Center, with Pratt and Whitney Rocketdyne and Aerojet as the prime contractors, and with testing conducted at the NASA Stennis Space Center.

The IPD was intended to demonstrate several new technologies, including high-performance, long-life technologies and materials for reuseable applications and a gas–gas injection MCC (for the first time). It is the first full flow staged-combustion (FFSC) cycle engine to be developed in the United States. In a FFSC engine, all of the propellants flow through oxygen-rich and fuel-rich preburners, through the turbopumps and into the main chamber, making 100% of the energy of the propellants available to produce thrust. Because all of the propellants flow through the turbines, it is possible to trade the increased mass flow for lower turbine temperatures, thus reducing the thermal stress on the turbine and increasing life and reliability. Also, since the hydrogen turbopump is powered by the hydrogen-rich preburner and the oxygen turbopump is powered by the oxygen-rich preburner, the catastrophic consequences of an interpropellant seal leak are avoided. The hydrogen-rich and oxygen-rich gases from the preburners are also used to pressurize their respective tanks, eliminating the need to use a heat exchanger and eliminating another catastrophic failure mode. Hydrostatic bearings in the oxygen and hydrogen turbopumps are another important technology demonstrated by the IPD. Hydrostatic bearings are supported by the fluid they pump (except during startup and shutdown), which simplifies manufacturability and virtually eliminates friction and wear.

The IPD is a ground-based demonstrator with a thrust of 250,000 lbf, mixture ratio 6, 2:1 throttle ratio, and a truncated nozzle. Components of the engine are designed for a lifetime of 200 missions.

B. Upper-Stage Engines

The Pratt and Whitney RL-10A and RL-10B family of upper-stage engines is now more than 40 years old, and although numerous upgrades have been incorporated over the life of the engine, much of the design is now outdated and significant improvements in performance and reliability could be achieved with a new design. Furthermore, the engine is the sole available candidate for both Atlas V and Delta VI, making the USAF totally dependent on this engine for all large payload launches.

Development and qualification of a new engine to replace the RL-10 would be expensive, but given the number of failures in recent years, it would seem to be worthwhile to develop an alternative for this dependence on a single-engine option. The RL-10 has evolved significantly over its 45 years of service. It began in 1961 with a vacuum thrust of approximately 15,000 lb (6804 kg) for the RL-10A-1. Throughout a series of modifications the thrust evolved to an average level of 24,750 lb (11,226 kg) in the RL-10B-2. Unfortunately this has reduced the margins somewhat, but has resulted in a very useful engine that has probably had every possible ounce of thrust rung out of it. Any further development will probably have to come from a new engine design.

Currently, the EELVs have only one upper stage, some version of the RL-10. The Delta IV of Boeing uses the RL-10B-2, while the Atlas V of Lockheed Martin uses the RL-10A-4-1 or -2. What this means is that both of the major launch systems in use today by the USAF and other DoD systems are dependent on the same second-stage engine. Should a failure occur that involves the second-stage engine, all launches with these engines would probably be frozen until the root cause was identified and corrected, which could take as much as a year or more. Although the probability of this is not high, nonetheless it is not zero. In a time of crisis, this could be extremely debilitating to the U.S. national defense needs, if neither of these systems were available for launch. Therefore it is in the nation's best interest to develop a second system for the upper stage of these vehicles.

Several organizations have recommended that a new second-stage engine ought to be developed in the thrust class of 50,000–100,000 lb (22,680–45,359 kg). Industry has responded to the perceived need with Pratt and Whitney developing the RL-60, Rocketdyne, the MB-60, Aerojet the AJ-60, and NGC also has an upper-stage engine technology (USET) funded program to develop a 40,000-lb (18,144-kg) LH_2 engine. All of these are in various stages of development. The MB-60 has components with a technology readiness level (TRL) between 6 and 9 depending on the component. The new RL-60 upper-stage engine (LO_2/LH_2) is under some level of development as Pratt and Whitney teamed with several international partners. Volvo is producing the nozzle while Ishikawajima–Harima Heavy Industry (IHI) is providing the hydrogen turbopump.

Aerojet has worked on the design of their AJ-60 concept but has yet to develop the hardware. They are, however, currently developing more physics-based models that are necessary to make real progress in a virtual engine design process. Such a process could have huge leverage in controlling the overall risks of implementing a full AJ-60 development.

All of these options offer various degrees of higher thrust, higher performance, and higher reliability capabilities than the RL-10 engine. If heavier payloads are to be placed into higher orbits, this additional capability will be needed, providing yet another reason to develop a new upper-stage engine for this application. The engine options that appear to be most suitable for the USAF and DoD missions are shown in Table 19.

The Atlas V and Delta IV programs both rely on a single-engine design for the upper stage (the Pratt and Whitney RL-10). This results in the risk of a single-point failure for both of these launch vehicles, and the potential to interrupt the ability to launch heavy payloads for as much as one to two years if such a failure should occur. A second and alternative engine design, fully tested and certified for operational readiness, is required to correct this deficiency. In addition, certain missions will likely require a higher-thrust level than that which is currently available, providing even more reason to develop a second engine option.

However, one must notice that most of the engine designs identified above are not in full-funded development. The development of a new rocket engine or motor is a very expensive proposition, costing between $100 million and $200 million, or even more if problems are encountered during development. In this time of few launches and a surfeit of competitive launch vehicles, the development of a new engine is a very risky business. As a result, because of the high cost and active competition, the rocket propulsion industry has been loath to invest the funds in what they view as a high-risk venture. The engine concepts above have many technology improvements designed in, but not validated. Many will probably not see much further development without significant government funding. That picture could become very different if the nNation commits to a serious well-funded long-term program for a new family of responsive space lift vehicles to support a major new total capability in-space architecture.

C. Strap-On Booster Technologies

One of the most effective ways to upgrade the payload capability of launch vehicles is to add strap-ons to the first stage. Solid strap-ons have been used frequently for this purpose, but liquids could also be employed. Some of the liquid and hybrid-propellant boosters currently being developed by the various FALCON program contractors are of an appropriate thrust level that could be a low-cost alternative to solids for this purpose. Both new solid booster technologies in work under IHPRPT, and possibly leveraging liquid and hybrid-propellant booster concepts that might be developed

Table 19 Upper-stage engine options

	Pratt & Whitney RL-10A-4	Pratt & Whitney RL-10	Rocketdyne MB-60	Northrop–Grumman TR-40	Aerojet AJ-60
Thrust, lbf (kgf) × 1000	23.3 (10.5)	60 (27.2)	60 (27.2)	40 (18)	60 (27.2)
I_{sp}, s	451	460	467		461
Mixture ratio	5.5	5.8	5.4		
Chamber pressure psia, kPa	620 (4275)	1250 (8618)	2000 (13,789)		1800 (12,410)
Weight, lb (kg)	375 (170)	1200 (544)	1300 (590)		
Area ratio	84	285	300		250
Cycle	Closed expander	Closed expander	Expander bleed	Split expander	Expander
Status	Production	Preliminary testing	Component testing	Paper	Paper

Table 20 Typical performance characteristics of monopropellant hydrazine thrusters

Thrust ranges	0.025–125 lbf (0.011–57 kgf)
I_{sp} range	225–239 lbf-s/lbm
Restart capability	750,000 starts >50 lbf (23 kgf)
Pressure operating range	350 psia blowdown <100 psia (0.69 kPa)
Radiative thermal control	Heaters, various types of heat shields and thermal shunts, as needed

for a new USAF space lift vehicle, should be studied for this application.

Lockheed Martin Space Systems has worked on hybrid propulsion technologies since 1989. Their initial studies were focused on replacing the solid rocket boosters on the space shuttle after the Challenger disaster. They worked with the American Rocket Company (AMROC) and during the DM-01, DM-02, and hybrid technology operational program (HyTOP) motor developments, which eventually led to the NASA/DARPA hybrid propulsion development program (HPDP). Within the HPDP, Lockheed Martin tested numerous technologies and increased the TRLs. Under the DARPA/USAF FALCON program, a number of tests were performed to demonstrate stable hybrid rocket performance. The largest hybrid motor tested to date using the staged-combustion system was the HPDP 250,000-lbf (113,398-kgf) motor, which had a ∼72-in. (2 m) diam and was 30 ft (9 m) long. The motor was tested 3 times for a total burn duration of 80 s. These tests demonstrated that the system could be successfully scaled to higher-thrust motors that could potentially be used for booster or first-stage applications.

VIII. Current Status of Spacecraft Onboard Propulsion Technology

A. Chemical Propulsion

Conventional chemical liquid-propellant propulsion systems in use today are either monopropellant or bipropellant. Liquid-bipropellant systems are higher performers, but are more complex. Monopropellant systems provide a single propellant for decomposition at the catalyst bed of the combustion chamber. Widely used, highly reliable, state-of-the art chemical systems are monopropellant hydrazine (N_2H_4) and bipropellant propulsion systems such as mixed oxides of nitrogen (MON) and monomethylhydrazine (MMH). For orbit circularization and station acquisition, dual-mode bipropellant systems using MON/N_2H_4 are also in use.

1. Monopropellant Engines

Hydrazine thrusters are typically used on satellites or other spacecraft to provide either altitude (angular) control or changes in linear velocity (ΔV). A system which provides attitude control is known as either ACS (attitude control system) or RCS. Some vehicles employ both ACS and ΔV thrusters. Generally, ACS thrusters tend to be low thrust ($\leq$ 5 lbf), and ΔV thrusters tend to be in the higher-thrust range (10–100 lbf). Key capabilities of monopropellant thrusters available in the U.S. are summarized in Table 20.

2. Bipropellant Engines

Key features and capabilities of bipropellant engines currently available in the U.S. are summarized in Tables 21 and 22. Radiative thermal control is generally used in this thrust class. An innovative bipropellant thruster designated as a secondary combustion augmented thruster (SCAT), has recently been flight qualified and flown. This thruster operates in the bipropellant mode on MON/N_2H_4 until the oxidizer is expended and then operates as a monopropellant thruster until all fuel is expended. The engine has a thrust range of 5–15 lbf (2.3–6.8 kgf) and is regeneratively cooled with a bipropellant I_{sp} max of 326 lbf-s/lbm. MON/MMH liquid apogee engines are typically used in combination with low-thrust MON/MMH thrusters used for on-orbit propulsive functions. MON/N_2H_4 liquid apogee engines are advantageous for spacecraft dual-mode propulsion systems that use monopropellant hydrazine catalytic thrusters for ACS and electrothermal hydrazine, or hydrazine arcjet, thrusters for on-orbit velocity control propulsive functions.

The manufacturers of monopropellant hydrazine and low- and high-thrust bipropellant engines in the U.S. are Aerojet, Redmond, WA; American Pacific, Niagara Falls, NY; and NGC Space Division, Redondo Beach, CA.

B. Electric Propulsion

1. Electric Powered Propulsion Systems

The expanding range of spacecraft size, power availability, and changes in the commercial spacecraft industry environment presented new challenges to the chemical propulsion community. There was a clear need to derive higher performance propellants and/or thrusters. The advent of power-rich spacecraft architectures

Table 21 MON/MMH bipropellant thrusters, low thrust

Thrust range	0.4–5 lbf (0.18–2.27 kgf)
I_{sp} range	250–295 lbf-s/lbm
Restart capability	Multiple
Pressure operating range	350 psia blowdown <100 psia (0.69 kPa)
Radiative thermal control	Heat shields, heaters, dog houses

Table 22 MON/MMH and MON/N_2H_4 bipropellant thrusters, high thrust (liquid apogee engines)

Thrust range	100–110 lbf (45.4–49.9 kgf)
I_{sp} range	305–326 lbf-s/lbm
Restart capability	Multiple
Engine inlet operating pressure	250 psia (1724 kPa)
Radiative/film thermal control	Heat shields, heaters, dog houses

provided the opportunity to take advantage of propulsion options that provide both high power and high I_{sp}. Reducing the onboard propulsion-system wet mass requirement can either decrease spacecraft mass or increase payload capability. In addition, greater demands can be placed on the onboard propulsion system, including increased repositioning or longer duration orbit maintenance—increasing useful life. Another option of reduced propulsion-system wet mass might be a step down to a lower weight-class launch vehicle. These performance enhancements targeted by commercial satellite owners are also desirable for military satellites. The propulsion industry accepted these challenges and was readily transitioned to electric propulsion where appropriate and useful. The extent to which the commercial satellite industry has embraced electric propulsion is well evident by the simple observation that there are about 200 operational satellites using electric-propulsion systems as of early 2006.

2. *Electrothermal Thrusters*

Starting with the implementation of the electrothermal hydrazine thruster first by TRW (NGC) on Intelsat V and then by RCA AstroElectronics (now Lockheed Martin) communication satellites, a significant I_{sp} improvement (~28%) from 225 to 295/300 s (0.43–0.50 kW power) was achieved with these thrusters, which electrically heat the decomposition products of monopropellant hydrazine to higher chamber temperatures. Without the complexity of carrying an oxidizer onboard, this I_{sp} is competitive with low-thrust bipropellant systems of 2.2–5 lbf (10–22 N), which provide an I_{sp} of 295/300 s

3. *Arcjet Thrusters*

In the early 1990s, Lockheed used the Aerojet MR-509 hydrazine arcjet system (1.8 kW power level, I_{sp} of 502 s on its series 7000 satellites). The arcjet continues to evolve with the latest Lockheed satellite bus, the A2100 satellites, which use the MR-510 arcjet system (2.2 kW, 582 s nominal I_{sp} thrusters) for north–south station keeping. This thruster takes advantage of the higher satellite power available to substantially increase performance while maintaining the simplicity of a single propellant for onboard propulsive functions.

4. *Ion Thrusters*

In the late 1990s, the Hughes Space Division (now Boeing Electrodynamics) successfully introduced the xenon ion propulsion system (XIPS), a xenon propellant girded ion thruster on its BSS 702 commercial communications GEO (geosynchronous Earth orbit) satellites. The thruster consists of a discharge hollow cathode, three-ring magnetic cusp confinement, three-grid accelerator, and neutralizer hollow cathode. The three-grid accelerator used in the 9.8 in. (25 cm) thruster uses shaped molybdenum grids with ~11, 000 apertures to produce the high-perveance (72 pervs at fail power) xenon ion beam. The XIPS 9.8-in. (25 cm) ion thrusters and the associated power supplies xenon power controller (XPCs) operate in two modes, 2.2 kW for typical on-orbit functions and 4.4 kW for augmenting orbit raising. The high-power mode uses ~4.5 kW of bus power to produce a 1.2-kV, 3-A ion beam. The thruster in this mode produces 165-mN thrust at an I_{sp} of ~3500 s. The high-power mode is used exclusively for the orbit insertion phase, which greatly reduces the amount of chemical propellant carried by the spacecraft for this task. Nearly continuous operation in the high-power mode for times of 500–1000 hr is required, depending on the launch vehicle and satellite weight. A low-power mode in which the thruster consumes about 2.2 kW of bus power is used for the station-keeping function. In the low-power mode, the beam acceleration voltage is kept the same and the discharge current and gas flow are reduced to generate a 1.2-kV, 1.43-A beam. In this mode, the thruster produces 79 mN of thrust. Because the beam voltage remains unchanged for the high-power mode and the thruster mass-utilization efficiency is nearly the same, the I_{sp} degrades only slightly compared to the high-power mode to ~3400 s. Typical performance parameters of the 9.8 in. (25-cm) thruster are summarized in Table 23. High confidence in the use of the XIPS was clearly established by the highly successful flight demonstration program of the N-STAR ion thruster on the "new millennium sponsored" deep space spacecraft.

Table 23 Typical parameters of the 25-cm XIPS thruster

	Low-power station keeping	High-power orbit raising
Active grid diameter, in. (cm)	9.8 (25)	25
Average I_{sp}, s	3400	3500
Thrust, lbf (mN)	0.02 (79)	0.04 (165)
Total power consumed, kW	2.2	4.5
Mass-utilization efficiency, %	80	82
Typical electrical efficiency, %	87	87

The state of the art on the XIPS electric-propulsion system is described in [3]. The Boeing 702 spacecraft has a chemical propulsion liquid apogee engine, but the use of the high-power mode of XIPS in the obit insertion phase greatly reduces the wet mass carried by the spacecraft for this task. The high I_{sp} of the ion engines for north–south station keeping is an additional large savings in propulsion-system wet mass over on-orbit systems, which use monopropellants or bipropellants for this function.

The military Gapfiller satellite, which has a flight date of late 2006, will use the BSS 702 9.8-in. (25-cm) version of XIPS. Aerojet also has a major development effort in the ion thruster system technology. Aerojet is completing the thruster, propellant management, and digital control system designs on the 0.5–7.5 kW NASA's evolutionary xenon thruster (NEXT) based on an improved version of the N-Star XIPS, led by the NASA Glenn Research Center. Boeing is developing the power processor. The NEXT system, when qualified, will provide significantly expanded capability for Discovery-class solar electric-propulsion missions. The 15.75-in. (40-cm) NEXT ion thruster will also be available for other spacecraft applications.

5. *Hall-Effect Thrusters*

The Russians, in 1971, flew the first Hall-effect thruster, or HET (sometimes identified as a stationary plasma thruster, SPT) on the METEOR spacecraft. Over the next two decades, several dozen 0.66-kW APT-70 thrusters were used operationally in space [4].

The use of Hall thrusters for satellite north–south station keeping promises great savings in wet mass over mono- or bipropellant chemical propulsion systems. An overview of the underlying physics involved in a Hall thruster is available in [5]. A typical propellant for a Hall thruster is a high molecular weight inert gas such as xenon. A power processor is used to generate an electrical discharge between a cathode and an annular anode through which the majority of propellant is injected. A critical element of the device is the incorporation of a radial magnetic field, which serves to impart a spin to the electrons coming from the cathode and to retard their flow to the anode. The spinning electrons collide with the neutral xenon, ionizing it. The xenon ions are then accelerated electostatically from the discharge chamber by the electric potential maintained across the electrodes by the power processor. The velocity of the exiting ions, and hence the I_{sp} is governed by the voltage applied to the discharge power supply and is typically 49,212–52,493 fps (15, 000–16, 000 m/s) at 300 V. The first flight of a Hall thruster on a U.S. spacecraft was in 1998 on STEX, a Naval Research Laboratory spacecraft. In 2004, Loral launched the MBSAT, a geosynchronous satellite, which uses four Faekel SPT-100 Hall thrusters for north–south station keeping. The performance characteristics of the SPT-100 class of thrusters are shown in Table 24.

Aerojet Redmond is developing and flight qualifying a 4.5-kW Hall thruster system for the Lockheed Martin build of the USAF advanced extra-high-frequency (EHF) satellites. These Hall thrusters will operate at two thrust levels, a high thrust for partial orbit transfer and lower thrust for station-keeping requirements. A launch date is projected as the fourth quarter of 2006.

Table 24 Characteristics of SPT-100 Hall-effect thrusters

Propellant	Xenon
Thrust, lbf (mN)	0.02 (80)
Power, kW	1.35
I_{sp}, s	1600
Efficiency, %	50
Life, h	>7000

The availability of flight-qualified, flight-proven ion and Hall thrusters can be expected to increase the manifesting of these technologies because of their high I_{sp}s when compared to chemical propulsion systems. Each type of electric powered thruster has its area of applicability. Ion engines can deliver higher I_{sp}s and are well suited to missions with high Delta V requirements. In addition to satellite north–south station keeping and partial orbit transfer requirements, Hall thrusters would be suitable for applications such as Earth transfer missions. In fact, the European Space Agency (ESA) SMART 1 mission has already successfully implemented this technology. SMART 1 is a small lunar orbiter, which was launched in September 2003 as an auxiliary payload on the Ariane 5. SMART 1 uses a 1.4-kW Hall thruster and reached the first moon orbit in December 2004. Because of the mass limitation of the spacecraft and the consequent limitation in the electrical power, the thruster used on SMART 1 is a scaled down version of the PPS-1350 thruster developed and qualified by SNMECA (France) for geosynchronous missions. SMART 1 used its thrusters in a variable power mode (450–1220 W) in this application, which serves as a benchmark for other ETO missions using electric propulsion.

To ensure broader application of ion or Hall thrusters, stronger emphasis needs to be placed on the development, produceability, and reliability enhancement of the components of the entire electric-propulsion subsystem, which includes not only the thruster but also the propellant feed system and the power processing unit (PPU). Two companies, Moog and Vacco Industries, are leading efforts to produce propellant-management components and systems for flight electric-propulsion systems. They are also developing next-generation designs that will trim the weight of the propellant-management system. Moog was the supplier of the xenon-propellant-management assembly that is flight operational on the Loral MBSAT. Currently Vacco provides the propellant-management system components for the BSS 702 XIPS system. Vacco is in the process of qualifying highly integrated xenon latch valve modules for the Lockheed Martin EHF spacecraft. Historically the PPU has been the dominant cost driver for electric-propulsion systems because of the requirement for heavy power converters and thermal management systems. Aerojet Redmond designs and builds high-power converters to support the electric-propulsion subsystems they manufacture. They are also working on development of solar-electric direct drive (i.e., using a high-voltage solar array to provide power directly to a Hall thruster at voltage levels needed to drive thruster discharge). Qualification of solar-electric direct drive would greatly reduce the cost and weight of a Hall electric-propulsion system while reducing array size. Reduction in array size is an added savings in spacecraft weight. The potential payoff for direct drive makes this is an extremely worthy goal to pursue. Additional weight savings could be obtained with direct drive using power from advanced solar arrays now available commercially, such as ENTECH/ABLE's solar concentrator arrays with refractive linear element technology. The latter type array provided the 2.7-kW power source for the successful deep space 1 basic mission and its extended mission to the comet Borrelly in 2001 [6]. ENTECH/ABLE also has available a next-generation array that is a stretched lens array. The synergy of coupling flight-proven advanced array technologies with direct drive for Hall thrusters needs to be explored.

To satisfy increasing demand, a larger industrial base is needed over what now exists for the manufacture of electric-propulsion systems and components. Boeing electrodynamics division and Aerojet Redmond appear to be the only commercial sources of girded ion thrusters and power converters. Aerojet is the only U.S. source that has ready-for-flight application Hall thruster hardware. In addition to the dual-thrust 4.5-kW Hall thruster under contract to Lockheed Martin for the EHF spacecraft, Aerojet has a 2.2-kW Hall thruster flight prototype unit fabricated and tested. Busek is developing low-watt Hall thrusters under IHPRPT. These are discussed under that heading. Excellent research and development on ion and Hall thrusters and power conditioning units are being conducted at NASA Glenn Research Center, but there needs to be more transfer of technology to companies who can build the product.

C. Micropropulsion Systems

The need for precise positioning of multiple small (micro) satellites or formation flying requires a propulsion system capable of delivering continuous microthrust levels. One thruster type suited for small-satellite, low-power applications that could be used for this function is Aerojet's pulse plasma thruster (PPT), designated PRS-1. This thruster was successfully flown on the EO-1 mission. PPTs rely on the Lorenz force generated by an arc passing from anode to cathode and the self-induced magnetic fields to accelerate a small quantity of chlorofluorocarbon propellant. Teflon has been used as the propellant to date. Pulsed electromagnetic thruster systems consist of the accelerating electrodes, energy storage unit, power conditioning unit, igniter supply, and propellant feed system.

During operation, the energy storage capacitor is first charged to between 1 and 2 kV, and the ignition supply is then activated to generate low-density plasma, which permits the energy storage capacitor to discharge across the face of the Teflon propellant bar. The peak current level is typically between 2 and 15 kA, and the arc duration is between 5 and 20 μs. The pulse cycle can be repeated at a rate compatible with the available spacecraft power. The propellant feed system consists of a negator spring, which pushes the solid Teflon bar against a stop on the anode electrode. The characteristics of the PRS-1 pulsed plasma system, flown on the NASA Goddard Earth Observing-1 (EO-1) spacecraft, are shown in Table 25.

The PPT-1 has demonstrated control of the spacecraft pitch with the momentum wheels completely disabled, including during image acquisition with the advanced land imager instrument. On-orbit tests have demonstrated no detectable electromagnetic interference with the spacecraft, spacecraft communications, or the payload instrument.

A micro-PPT was developed to provide attitude control on FalconSat when it is launched. The characteristics of this thruster are shown in Table 26. Colloid thruster technology also received considerable attention. Colloid thrusters have demonstrated remarkable range, precision, and controllability in the micro-newton regime, with single spray sources providing thrust in the 0.3–3 μN range with nano-newton resolution. The response time across the full thrust range is only 2–3 s, achieved via a microvalve, and fine adjustments, performed by varying acceleration voltages, may be realized in milliseconds. The characteristics of the microthrusters for the NASA ST7 mission are shown in Table 26.

IX. Current Status of In-Space Propulsion Technology

Current and future needs for satellites and in-space vehicles exist in the following areas:

1) Strategic assets for communication, early warning, Earth observation, navigation, reconnaissance, surveillance, and weather;

2) technology development work in space;

3) responsive space operations.

All of the U.S. satellites and technology platforms require propulsion subsystems operating in space to provide the impulse necessary to adjust velocity, change orbit altitude, provide attitude control, station keeping, and end-of-life deorbit. These propulsion needs are being satisfied currently by using state-of-the-art chemical propulsion and increasingly, by electric-propulsion subsystems. It should be noted that the "state of the art" has been undergoing major changes over the past 15 years, and therefore represents a very advanced capability in many areas. For kilogram-weight-class technology development satellites with unique mission capabilities, micropropulsion systems may be required for all maneuvers other

Table 25 PPT performance characteristics [7,8]

Characteristic	EO-1	Dawgstar
Maximum input power	70 W (one thruster-EO-1 operations)-100 W design	15.6 W (two thrusters; slow charge) 36 W (two thrusters; fast charge)
Thrusters/system	2	8
Total system impulse	405 lbf-s (1850 N-s) (EO-1 propel. load) >3372 lbf-s (15,000 N-s) (system life)	>337 lbf-s (1500 N-s)
Impulse bit	20–193 mlbf-s (90–860 mN-s), throttleable	15 mlbf-s (66 mN-s)
Pulse energy	6.3–41 ft-lb (8.5–56 Joules), throttleable	3.6 ft-lb (4.9 Joules)
Maximum thrust	193 lbf (860 N) (EO-1); 0.3 mlbf (1.2 mN) (design)	45 mlbf (200 mN) (high-speed mode)
I_{sp}	650–1,350 s	$332 + 40$ s
Thrust to power ratio	2.74 mlbf/W (12.3 mN/W)	2 mlbf/W (9.7 mN/W)
Total mass	10.8 lb (4.9 kg) (2 PPTs, a power processing unit, and propellant)	9.2 lb (4.2 kg) (8 PPTs, a power processing unit, and propellant)
Propellant	PTFE	PTFE
Propellant mass	0.15 lb (0.07 kg)/thruster (as fueled)	0.15 lb (0.07 kg)/thruster; 1.2 lb (0.56 kg)/system

Table 26 Summary of small electric thruster capabilities

	Colloid thruster	Micropulsed plasma thruster
Dry mass	6.6 lb (3 kg)	1.54 lb (0.7 kg)
I_{sp}	1000 s	800 s
Min. Ibit	0.022 lbf-s (0.1 N-s)	16.9 lbf (75 N)
Power	25 W	10 W
Delta V	181 fps (360 m/s)	1230 fps (375 m/s)
TRL	7	7
Demo(s)	NASA ST7	FalconSat–3

than rapid inclination change. A summary of the in-space rocket engines and thrusters in use today has already been presented in Table 1.

The nation will continue to need higher performance in-space propulsion technologies for military satellites, commercial space systems, and civil space vehicles of various types for various missions. There are a number of approaches for meeting these needs. For large Earth orbiting assets, for example, one could use an onboard, low-thrust very high fuel efficiency performance system such as a Hall-effect thruster that would fire continuously to complete a large station change or repositioning maneuver at high I_{sp} (2000–3000 s). However, such maneuvers for large assets are measured in velocity changes of feet per second per hour and take weeks to travel 10,000 mile (16,093 km).

A second way would be to use a moderate-thrust (100–200 lbf (45–91 kgf)), modest-performance chemical propulsion thruster at 350–360 s, such as LO_2/N_2H_4 using a cryocooler to keep the LO_2 from boiling away. Velocity changes of hundreds of feet per second depending on the propellant mass available can be achieved in minutes to hours for platforms of thousands of pounds, permitting position changes of thousands of miles per day. With higher-thrust levels maneuvering times could be quite rapid.

One important technology that would permit multiple and longer life maneuvers of critical assets would be to use an on-orbit refueling system to resupply the propellants while changing stations. The on-orbit refueling capability would enable the space asset to stay alive as long as everything kept working and to make as many rapid station changes as required. The capability for autonomous, robotic on-orbit docking and refueling will be demonstrated with hydrazine and high-pressure helium in space by the end of 2006, by the DARDA, USAF, NASA sponsored orbital express (OE).

A third way to implement rapid station changes would be to have a large (even on-orbit refuelable) space tug with high-performance electric propulsion for slow strategic moves or a high-thrust, modest-performance chemical system for responsive maneuvers. Or have some combination of propulsion systems on the tug that would fly up, dock with a key space asset, and move it to the desired new operational location. The tug could then demate from the spacecraft and fly on to reposition other assets as needed. This approach would entail a small fleet of permanently based space maneuvering tugs that would have no other function [with its own dedicated guidance, navigation, and control (GNC) and telemetry/command system] than to rapidly maneuver space assets to new stations as required. This fleet of tugs should also be on-orbit refuelable to extend their lifetimes.

Of course, some combination of the above approaches for slow or rapid maneuvering and repositioning could also be adopted for operational use to provide more robust, flexible, survivable, and long-life capabilities.

A. Chemical Propulsion Work Under IHPRPT

As discussed earlier, the USAF has established a program referred to as IHPRPT. This program is a joint government and industry effort focused on developing technologies for military global reach capability, strategic missiles, long-life or spacecraft capability, and tactical missile capability. The performance metric increases desired for spacecraft propulsion are shown in Table 27.

1. *Alternative Propellant Developments for Liquid-Propellant Engines*

Research and development is under way in house at AFRL and in industry on several energetic monopropellants for potential use for orbit circularization, orbit altitude/position changes, fly-out and maneuvering, and attitude control to meet the phase II IPRPHT goals (Table 28). Hydroxy ammonium nitrate (HAN) and AF-315E are two of the propellants under study. The main advantages are higher-density impulse than state-of-the-art chemical monopropellant or bipropellant systems and lower propellant toxicity. The industry is experimenting with various blends to achieve a balance between safety of handling and performance. Phase II potential has not yet been realized at the thruster level. Theoretical performance calculations indicate an I_{sp} of 250–270 s. Measured I_{sp} to date is -10% below that level. There are at least three leading technical challenges associated with HAN-type propellants: 1) selection of suitable chamber materials for the thrusters because the high-performance blends tend to run hotter than monopropellant hydrazine, 2) the high temperature needed on the catalyst bed for initial startup, and 3) ignition delay problems. These challenges are solvable, but the feasibility of large-scale propellant production and long-term material compatibility are still outstanding. These "designer" monopropellants may fill a small-niche mission need, such as low delta V, small spacecraft applications; however, existing state-of-the-art bipropellant systems are competitive in performance with HAN, and MON/MMH systems bring with them an already substantial history of proven performance and reliability.

2. *Combination Thrusters Dual-Mode Capability*

SCAT is the first thruster designed to operate with "bimodal" capability, in either a bipropellant or a monopropellant mode. In its monopropellant mode, SCAT decomposes hydrazine in a catalyst

Table 27 Performance metric increases for spacecraft propulsion

Spacecraft propulsion	Phase I	Phase II	Phase III
Improve $I_{tot}/mass_{(wet)}$ (electrostatic/electromagnetic)	20%/200%	35%/500%	75%/1250%
Improve I_{sp} (bipropellant/solar thermal)	5%/10%	10%/15%	20%/20%
Improve density-I_{sp} (monopropellant)	30%	50%	70%
Improve mass fraction (solar thermal)	15%	25%	35%

Table 28 Chemical propulsion IPRPHT phase II goals.

Activities	Objective
Phase II demonstration	To demonstrate technologies that enable realization of USAF IHRPT phase II goals with improved I_{sp}, life, thrust to weight and lower cost
Liquid engine alternative propellant development program (LEAP-DP)	1) To develop and demonstrate a catalytic engine for AF-M315E monopropellant that delivers IHPRPT phase II performance
	2) To design and fabricate a 25-lbf heavyweight workhorse thruster and flight weight thrusters
	3) To hot-fire demonstrate flight weight thruster with AF-M315E monopropellant
Energetic propellant	To develop and demonstrate the scale up of energetic propellant formulations

bed chamber [9]. The decomposition products ($NH_3 + N_2 + H_2$) flow out through a second small chamber and exit through a conventional nozzle with an expansion ratio of about 100:1. The conventional monopropellant N_2H_4 thruster has a steady state I_{sp} of $\sim$230 s and can provide thrusts from 0.8 to 4.5 lbf (0.36–2 kgf). This system has been under development for years at Northrop-Grumman Corporation and is now flight qualified on several NGC satellites.

In its bipropellant mode, N_2H_4 is turned on to cool the second chamber, vaporizes, and then combusts the NO_2 vapor with the N_2H_4 decomposition products in the second chamber to produce an I_{sp} of $\sim$325 s. Because the second chamber is regen cooled, it can be made of nonrefractory metals such as nickel and L-605 alloy. This provides essentially unlimited operating life. In its bipropellant mode, the thruster can produce 4–14 lbf (1.8–6.35 kgf) thrust. NGC is developing higher-thrust versions and also exploring other propellant combinations.

SCAT'S combination of operating modes permits the most efficient use of onboard propellant, thereby providing greater mission flexibility. SCAT allows a single engine to burn either hydrazine (monopropellant mode) or hydrazine and N_2O_4 (bipropellant mode), which means that the oxidizer tank can be fully depleted and then the fuel tank can be fully depleted. This total utilization of available propellant is not possible with any other bipropellant thruster. SCAT also has a very wide range of allowable mixture ratios (0.95–1.6), so it becomes much easier to balance oxidizer-versus-fuel usage over a mission, unlike common LAEs, which have very tight limits on operating mixture ratio (e.g., 1 ± 0.05). It can also operate over a deep or wide blowdown range, 4.5–1, just like catalytic monopropellant hydrazine thrusters.

B. On-Orbit Refueling: Orbital Express

DARPA is sponsoring the orbital express spacecraft program to demonstrate the practicality of carrying out autonomous/robotic on-orbit refueling of spacecraft propulsion systems. Reasons for the DARPA interest in in-space refueling is that some satellites that retain full functionality have to be retired because the original propellant load carried into orbit with the satellite is exhausted. If spacecraft can be designed to be refueled in space, they could continue to operate for much greater periods. To date, the United States has not been successful in performing acceptable robotic transfer of fuels in space. The capabilitites to be demonstrated by the orbital express include the following:

1) autonomous rendezvous and docking of two independent spacecrafts;

2) an efficient zero-g propellant transfer pump;

3) accomplishing the transfer with minimal venting of propellants and pressurant gases;

4) meeting a requirement for zero dribble from quick disconnect fittings;

5) reuse of pressurant gases;

6) completely autonomous operations;

7) replacement of deficient electrical boxes/subsystems.

A primary requirement is to have no spacecraft contamination occur during or following the transfer. The present plan is to demonstrate transfer of hydrazine only. Transition to a fully operational fluid transfer docking satellite and autonomous servicing system could have important benefits for mobility of future USAF and NASA satellite constellations and for reusable fly-out tugs for various in-space missions. Some assets could be launched into orbit without having to have an onboard propellant capacity to support all of the missions the space vehicle may be functionally capable of carrying out.

X. Potential Future Propulsion Needs for Space Exploration

Enabling the president's vision to "extend human presence across the solar system, starting with a human return to the moon by the year 2020, in preparation for human exploration of Mars and other destinations" will require a robust and reliable fleet of launch vehicles [10]. With the retirement of the shuttle scheduled for the year 2010, the United States is now faced with the challenge of replacing its only human-rated and heavy-lift launch vehicle. To meet this challenge, NASA was elected to pursue an evolutionary shuttle-derived approach to ensure the rapid development of a launch vehicle for the robotic lunar missions targeted for 2008 and the human lunar missions targeted for 2015–2020 (i.e., the crew launch vehicle, or CLV). Adding to the difficulty of maintaining U.S. technical superiority is the new emergence of China as the third nation to successfully put a human into space. The United States will now have to compete with multiple nations in its quest to remain the world leader in space exploration.

When NASA initiated the Apollo program in the 1960s, the Saturn V was developed with the capability to lift 140 t to LEO. This vehicle provided the heavy-lift capability that enabled NASA to complete lunar missions with a single launch. The most recent human exploration concept studies indicate that future missions will require payloads in excess of 100 t to LEO. Currently, the largest payload capability provided by an existing launch vehicle is the space shuttle at 27.5 t. Although shuttle-derived vehicle concepts appear to provide an expedient path to the development of a heavy-lift launch vehicle, initial vehicle concept studies indicate that expendable or partially expendable systems will provide the most cost-effective access to space for human exploration missions. The largest Delta- and Atlas-class heavy expendable launchers can only deliver payloads in the 20- to 25-t range. Based on economic factors and the limited number of planned exploration missions, the United States may have to delay the development of a large heavy-lift launch system, which could force NASA to rely on the use of multiple ETO launches with on-orbit assembly of the spacecraft and transfer of expendables performed in LEO.

The selection and sizing of the launch-vehicle fleet will represent a landmark architectural decision in the future exploration of space. The CONOPS developed for the lunar and follow-on Mars campaigns will dictate the design of each element required for these missions. The sizing of the TLI stage, the trans-Earth injection (TEI) stage, and 29 other vehicle subsystems in the notional architecture shown in Fig. 13 will be some of the primary drivers used to determine the optimum payload size for future launch vehicles.

Based on the most recent vehicle trades, it is not feasible, under the current NASA budget constraints, to pursue a single-launch approach for the deployment of a planetary exploration vehicle. A multilaunch approach will be required to enable human exploration of the moon, Mars, and other planets. The need to assemble an exploration spacecraft in LEO will necessitate accelerating the development of advanced autonomous rendezvous and docking (ARD) systems. These systems will provide NASA with the capability to launch unmanned elements (including propellant) of the exploration vehicle separately and then execute on-orbit assembly and propellant-loading operations in LEO. Advanced ARD systems will allow NASA to functional test the fully assembled exploration vehicle before launching the crew, which will enhance crew safety and provide time to work around any anomalous conditions that may occur during the vehicle checkout phase. Advanced ARD systems could also be used for station keeping in the event of a component or subsystem malfunction where replacement units would have to be delivered on a subsequent launch.

The development of highly efficient in-space propellant-management systems will provide NASA with the capability for long-term on-orbit cryogenic storage of propellants and autonomous transfer of stored propellant for in-space refueling operations. The development of propellant depots would provide extended service life to all reusable in-space propulsion systems and provide NASA with the option to launch transportation elements without fuel, providing additional payload mass margin. Along these lines, introduction of oxygen (and possibly fuel) production from in situ resources can have a significant impact on reducing mission cost and mass. For every kilogram delivered to the surface of the moon and Mars equates to approximately 8.8 lb (4 kg), or more, in LEO. Therefore, in situ propellant production (ISPP) of oxygen, which can make up 75% of propellant mass, can significantly reduce mission cost and mass. Full propellant production has an even greater impact, and ISPP also enables reusability of lander assets and can support in-space propellant depots and reusable in-space stages for reduced cost trans-Earth transportation.

Using its design approach, the Apollo program developed 10 different engine types specifically designed for each transportation element on the Saturn V. Although these same transportation elements and more will be required for future planetary exploration missions, it is not economically feasible to design, test, and build new unique engines for each element. As discussed previously, the only U.S.-built liquid-propellant upper-stage-enabling rocket engine is the RL-10XX oxygen/hydrogen engine, which was initially developed in the late 1950s. Even though the RL-10XX has gone through several design modifications to upgrade engine performance, an advanced upper-stage engine with multiple application capabilities is still identified as a key technology needed for its many applications in the human exploration of the moon and Mars.

NASA will conduct the studies and develop the architectures from which will come the requirements for launch systems, orbit transfer and trajectory insertion vehicles, in-space fluid management systems, and the many other propulsion elements needed for human exploration of the moon and Mars. Some of these requirements will be met from work now being done by NASA and the USAF, but many of them will require new systems. As we prepare for human space exploration, we must recall that the United States no longer has the base of industrial capability on which Apollo was built. Instead, the exploration program will work with an industry shrunk through unrelenting consolidation and losing its more experienced individuals as they age. If the exploration program is to succeed, its studies and plans will consider how best to invest both its advanced technology funds and its development funds so as to maximize the yield from, and improve, the capability that remains.

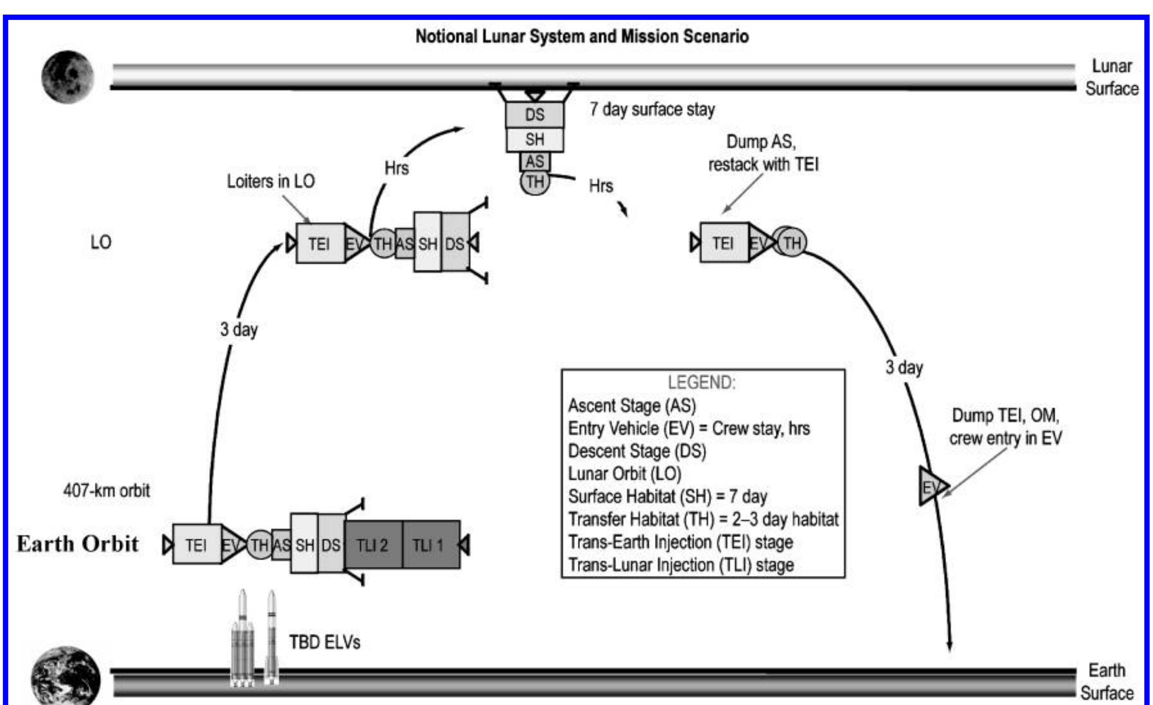


Fig. 13 Notional lunar system and mission scenario.

XI. Conclusions

Robotic and human space missions into and beyond LEO are very complex and risky endeavors. When humans are the primary cargo, especially to more distant locations such as the surface of the moon, all space flight operations become much more complex; more important there is now the element of great danger that must be addressed and reconciled. This statement is based on fact, not opinion and politics, and has been demonstrated in many ways over the history of space flight worldwide since the middle of the last century. In the case of human space flight, the subject of safety has to become the dominant factor and criterion for mission success of any kind. The cost of failure in space is enormous. The physics involved and the operational environment are extremely marginal and unforgiving.

Space flight is nothing like airplane/airline flight. In the first place, airplanes only fly up to 20–50,000 ft (6–15,240 m) and then, using the existing atmosphere or some gentle type of thrust reversal (or a combination), gradually reduce the energy required to fly from point to point inside the Earth's atmosphere using aerodynamic braking to allow a safe and highly controlled gentle landing. Space vehicles, however, must add a minimum of 30,000 fps (9144 m/s) of energy just to reach a minimum safe parking orbit, above most of the drag of the Earth's atmosphere. The energy required to go much further in the solar system (e.g., to the moon and beyond) can be 2–3 times that of ETO, plus the original 30,000 fps (9144 m/s).

Then, to ensure a gentle landing on the destination planet and/or back on Earth, all of this energy that was added by extremely powerful rockets must be reduced back down to nearly zero, relying upon frictional heating at hypersonic reentry speeds, greater than Mach 25 at temperatures greater than 5432°F (3000°C), and then drag to kill off the remaining subsonic velocity with aerodynamics. This is because all space vehicles (from the surface of the Earth to anywhere in space) are always extremely weight limited, even with the best rocket technology that could be developed in the near future. The cost, in terms of GLOW, to carry the full allotment of thermally (high thrust-to-weight) energized propulsion/propellant for all "safe braking" functions would be enormous, and the amount of payload would be prohibitively small (if not negative).

During powered space flight using rocket propulsion, because of the enormous amount of energy that must be released under extremely controlled, but nearly marginal and very high-stress conditions, it is this high-power/high-energy release period that is the most dangerous phase (or phases) of the mission. To reinforce this point it is important to note that the probability of a catastrophic airplane failure is about 1 in 10^7 flights, whereas for the space shuttle, the probability of catastrophic flight during the powered phase is about 4 in 10^2 flights, or about 1 in 100 flights if you consider actual results to date.

It is also interesting to note that from a power plant internal operation point of view, the horsepower-to-weight ratio of the best jet engine flying (either the F-22 or the Joint Strike Fighter) is about 19:1. This compares with an average horsepower-to-weight ratio for the 400,000-lbf (181,437 kgf) thrust SSME of about 800:1; and the initial start transient for the SSME takes the combustion chamber from about -400 to 7000°F (-240 to 3817°C) (in the core) in $\sim$1 s. Therefore, it is the rocket propulsion engine system that is the most critically stressed element associated with space flight, particularly human space flight. Yet, just when the nation's need is growing, the national capability on which it can draw is shrinking. Only the space-exploration program can address this problem and is given a charge whose execution requires that the problem be addressed.

To ensure mission success for future exploration missions, the nation must take strong definitive steps to retain and enhance today's diminishing propulsion technology knowledge and skill base. A concerted effort by the government to increase the level of much-needed new propulsion research and development activities would lead to improved performance and an overall reliability increase of future propulsion-system designs, as well as build on the essential lessons that were learned from propulsion systems and associated successes that were flown on actual missions during the earlier ICBM, Apollo, and STS eras.

Acknowledgments

Special thanks go to G. W. Elverum, Jr. (TRW/ATD, V.P., retired, now private consultant); Y. Brill (private consultant), G. Dressler (Northrop Grumman Space Technology, Chief Engineer for Propulsion) for technical support, and Aerojet, Pratt and Whitney Rocketdyne, and Northrop Grumman Space Technology for the use of their openly published marketing data and pictures available from their respective Web sites. Also thanks to John Wiley and Andy Gamble (NASA Marshall Space Flight Center) and the people at the 10 NASA field centers and headquarters who participated in the space propulsion strategic partnership planning efforts and provided significant background information for this document.

References

[1] Mueller, G. E., "'Apollo' the First Space Systems of Systems," *Human Exploration Space Technology Conference*, Dec. 2004.
[2] Sackheim, R. L., London, J. R., III, and Weeks, D. J., "The Future for Small Launch Vehicles," *Space Technology and Applications International Forum (STAIF–2005)*, Albuquerque, NM, Feb. 2005.
[3] Goebel, D. M., Martinez-Lavin, M., Bond, T. A., and King, A. M., "Performance of XIPS Electric Propulsion in On-Orbit Station Keeping of the Boeing 702 Spacecraft," AIAA Paper 2002-4348, July 2002.
[4] Bober, A. S., and Maslennikov, N. A., "SPT in Russia—New Achievements," *Proceedings of the 24th International Electric Propulsion Conference*, Moscow, Sept. 1995, pp. 54–60.
[5] Kaufman, H. R., "Technology of Closed-Drift Thrusters," *AIAA Journal*, Vol. 23, No. 1, Jan. 1985, pp. 78–87.
[6] Jones, P. A., Murphy, D. M., Allen, D. M., Caveny, L. H., and Piszczor, M. F., "SCARLET: A High-Payoff, Near Term Concentrator Solar Array," AIAA Paper 96-1021, Feb. 1996.
[7] Benson, S. W., Arrington, L. A., and Hoskins, W. A., "Development of PPT for EO-1 Spacecraft," AIAA Paper 99-2276, June 1999.
[8] Rayburn, C., Campbell, M., Hoskins, W. A., and Cassady, R. J., "Development of a Micro Pulsed Plasma Thruster for the Dawgstar Nanosatellite," AIAA Paper 2000-3256, July 2000.
[9] Sackheim, R. L., Hook, D. L., and Joseph, G. W., "Satellite Propulsion and Power System," Patent No. 5,572,865, filed 12 Nov. 1996.
[10] Bush, G. W., published vision statement, 14 Jan. 2004.

D. Talley
Associate Editor

This article has been cited by:

1. Akira Kakami, Shinji Beppu, Muneyuki Maiguma, Takeshi Tachibana. 2012. Performance of Dimethyl Ether Arcjet Thrusters and Its Dependence on Electrode Configurations. *Journal of Propulsion and Power* **28**:3, 603-608. [CrossRef]
2. Akira KakamiShinji BeppuMuneyuki MaigumaTakeshi Tachibana. 2012. Performance of Dimethyl Ether Arcjet Thrusters and its Dependence on Electrode Configurations. *Journal of Propulsion and Power* **28**:3, 603-608. [Citation] [PDF] [PDF Plus]
3. Laurent Catoire, Steven D. Chambreau, Ghanshyam L. Vaghjiani. 2012. Chemical kinetics interpretation of hypergolicity of dicyanamide ionic liquid-based systems. *Combustion and Flame* **159**:4, 1759-1768. [CrossRef]
4. Anthony B. Murphy. 2012. Transport coefficients of plasmas in mixtures of nitrogen and hydrogen. *Chemical Physics* **398**, 64-72. [CrossRef]
5. Akira KAKAMI, Shinji BEPPU, Muneyuki MAIGUMA, Takeshi TACHIBANA. 2012. Thrust Evaluation of an Arcjet Thruster Using Dimethyl Ether as a Propellant. *TRANSACTIONS OF THE JAPAN SOCIETY FOR AERONAUTICAL AND SPACE SCIENCES* **55**:2, 116-122. [CrossRef]
6. Richard Strunz, Jeffrey W. Herrmann. 2011. Reliability as an Independent Variable Applied to Liquid Rocket Engine Test Plans. *Journal of Propulsion and Power* **27**:5, 1032-1044. [CrossRef]
7. Richard StrunzJeffrey W. Herrmann. 2011. Reliability as an Independent Variable Applied to Liquid Rocket Engine Hot Fire Test Plans. *Journal of Propulsion and Power* **27**:5, 1032-1044. [Citation] [PDF] [PDF Plus]
8. 2011. Full issue in PDF / Numéro complet en form PDF. *Canadian Aeronautics and Space Journal* **57**:2, 115-153. [CrossRef]
9. Hanafy M. Omar, Moumen Idres, Raed Kafafy. 2011. Particle swarm optimization of a micro air-launch vehicle trajectory. *Canadian Aeronautics and Space Journal* **57**:2, 115-120. [CrossRef]
10. Hai-Bin TangXin-Ai ZhangYu LiuHai-Xing WangChen-Bo ShiBin Cai. 2011. Experimental Study of Startup Characteristics and Performance of Low-Power Arcjet. *Journal of Propulsion and Power* **27**:1, 218-226. [Citation] [PDF] [PDF Plus]
11. Ralph L. McNuttScience Satellites: Interplanetary Spacecraft . [CrossRef]
12. Hai-Xing Wang, Jin-Yue Geng, Xi Chen, Wen Xia Pan, A. B. Murphy. 2010. Modeling Study on the Flow, Heat Transfer and Energy Conversion Characteristics of Low-Power Arc-Heated Hydrogen/Nitrogen Thrusters. *Plasma Chemistry and Plasma Processing* **30**:6, 707-731. [CrossRef]
13. Xi Chen. 2010. The impact force acting on a flat plate exposed normally to a rarefied plasma plume issuing from an annular or circular nozzle. *Journal of Physics D: Applied Physics* **43**:31, 315205. [CrossRef]