Liquid Rocket Hydrocarbon Booster Engines (LRHCBE's) for Launch Vehicles – A Status Report

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Abstract

There have been very few new rocket engines fully developed and flown in the United States (US) in the last five decades. Since the end of the Space Shuttle Program and flight certification of the Delta IV and Atlas V Evolved Expendable Launch Vehicles (EELVs), the US Government has not seen its way clear to directly fund any completely new liquid rocket engine (LRE) development programs all the way through to flight qualification. There have been, however, some elements of new LRE development efforts, including advanced modeling and analysis and some limited testing of newer prototype LRE hardware. These have been sponsored and supported to some degree by the National Aeronautics and Space Administration (NASA) and the United States Air Force (USAF) in what have been thus far, separate projects. There also have been the development and flight qualification of two new, but completely funded with private, commercial resources, liquid hydrocarbon engines for the SpaceX Falcon family of commercial launch vehicles. This paper will discuss some of the more recent LRHCBE technical activities and briefly describe some of the technology development efforts for new LRHCBE's for both the US Government and some of the commercial low cost access to space, entrepreneurial organizations.

1.0 Introduction

In general, other than earth storable hypergolics (e.g., Titan II, which are no longer used for environmental impact reasons), there are two basic propellant options for a liquid propellant first stage (booster stage) of a launch vehicle. These are, of course, liquid oxygen and liquid hydrogen (LO_2/LH_2) or liquid oxygen and liquid hydrocarbon fuel (usually RP-1). Each of these primary liquid options has two sub choices for their respective power cycle, either open or closed for a total of four basic possibilities. Of course, there are also solid rocket options for either strap-on thrust augmentation stages or as the primary propulsion system for a launch vehicle booster stage. Each of these 1st stage solid rocket options is believed to have certain limitations for these purposes, which are probably quite controversial in some propulsion communities. However, it is not the purpose of this paper to compare the relative attributes and/or shortcomings of liquids versus solid propellant rockets for booster stage applications (which has been a long term historically controversial subject for the last 6 or 7 decades), but rather to focus on the status of development and use of liquid hydrocarbon engines for booster stages primarily in the U.S., but in some other countries (outside the U.S) national space programs as well.

2.0 Discussion of Hydrocarbon versus Liquid Hydrogen Fuel Booster Engines and Associated Power Cycles

Returning to liquid propellant booster rocket engines, a discussion of each of the four options will be presented next. Two of the four options for the booster stage main propulsion system (MPS) use a liquid hydrogen fueled rocket engine that is powered with either an open or gas generator cycle (option 1) as was used for the second and third stages of the Saturn V launch vehicle for the U.S. Apollo program or the RS-68 for the first stage of the Delta IV launch vehicle or a closed cycle (option 2) liquid hydrogen (LH_2) fueled rocket engine. There are really two further sub divisions within the closed LH_2 cycle engine category. These are 2-A, the fuel rich staged combustion cycle, as was used on the Space Shuttle Main Engine (SSME) or the expander cycle such as the RL-10 that is still being used in some version for all current U.S. launch vehicle second/upper liquid propellant stages, essentially some variation of the Centaur upper stage. Each of these options will be discussed briefly, below.

As shown in the simplified schematic in Figures 1-d and 2-e, a fuel rich staged combustion engine uses a high pressure pre-burner to generate hot gas to drive the main turbine for the engine turbo pumps and then injects this fuel rich hot gas exhaust from this turbine into the main combustion chamber (MCC) to mix and combust with liquid oxygen to generate engine thrust. In this cycle, all of the hot fuel rich gas is burned in the main combustor so none of the fuel is exhausted overboard without complete combustion, as is the case with the open gas generator cycle. This closed staged combustion engine cycle is generally 5 to 10% more efficient than a gas generator driven pump assembly engine. Before entering the pre-burner, all of the LH_2 fuel passes through the MCC cooling jacket to cool the chamber as shown in Figure 1-D. The other closed engine cycle, 2-B is designated as an expander cycle and has only been used with hydrogen fuel because of its very high vapor pressure. In this engine design, the hydrogen passes through the cooling jacket of the main combustion chamber (MCC) and exits the cooling jacket in a vaporized state to drive the turbopump assembly (TPA).

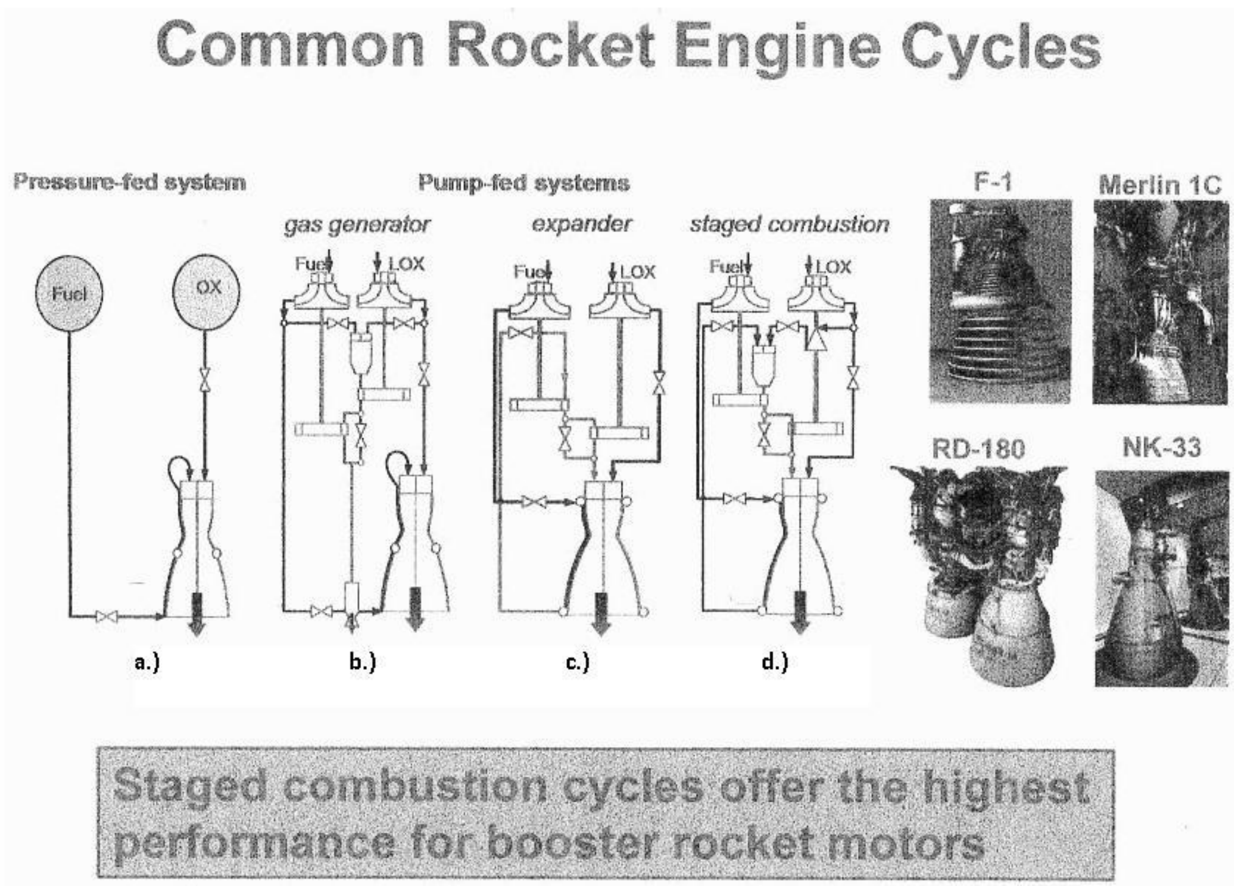

Figure 1: Common rocket engine cycles

This gaseous fuel is then expanded through the pump turbine to drive the pump assembly. The vaporized fuel from the turbine exhaust is injected back into the MCC to mix with the LO_2 in a liquid-gas combustion process to produce thrust. This is the cycle used for the RL-10 family of upper stage engines that has been in use for the last 50 years or so. A simplified schematic diagram of the expander cycle engine is shown in Figure 1-C. The engine (RL-10) is started up in a boot strap mode using H_2 tank ullage head pressure from the LH_2 tank to initially spin up the turbine. A LO_2/LH_2 booster stage main propulsion system (MCC) for a launch vehicle will result in a much higher specific impulse or I_{sp} (the highest chemical I_{sp} achieved is typically 390 to 410 seconds at sea level) than a hydrocarbon fueled engine. Use of a cryogenic liquid hydrogen (LH_2) fueled engine however, has some serious drawbacks for use in a booster first stage, because of its extremely low propellant density and its deep cryogenic properties, which requires hydrogen fuel to be stored at minus 420°F.

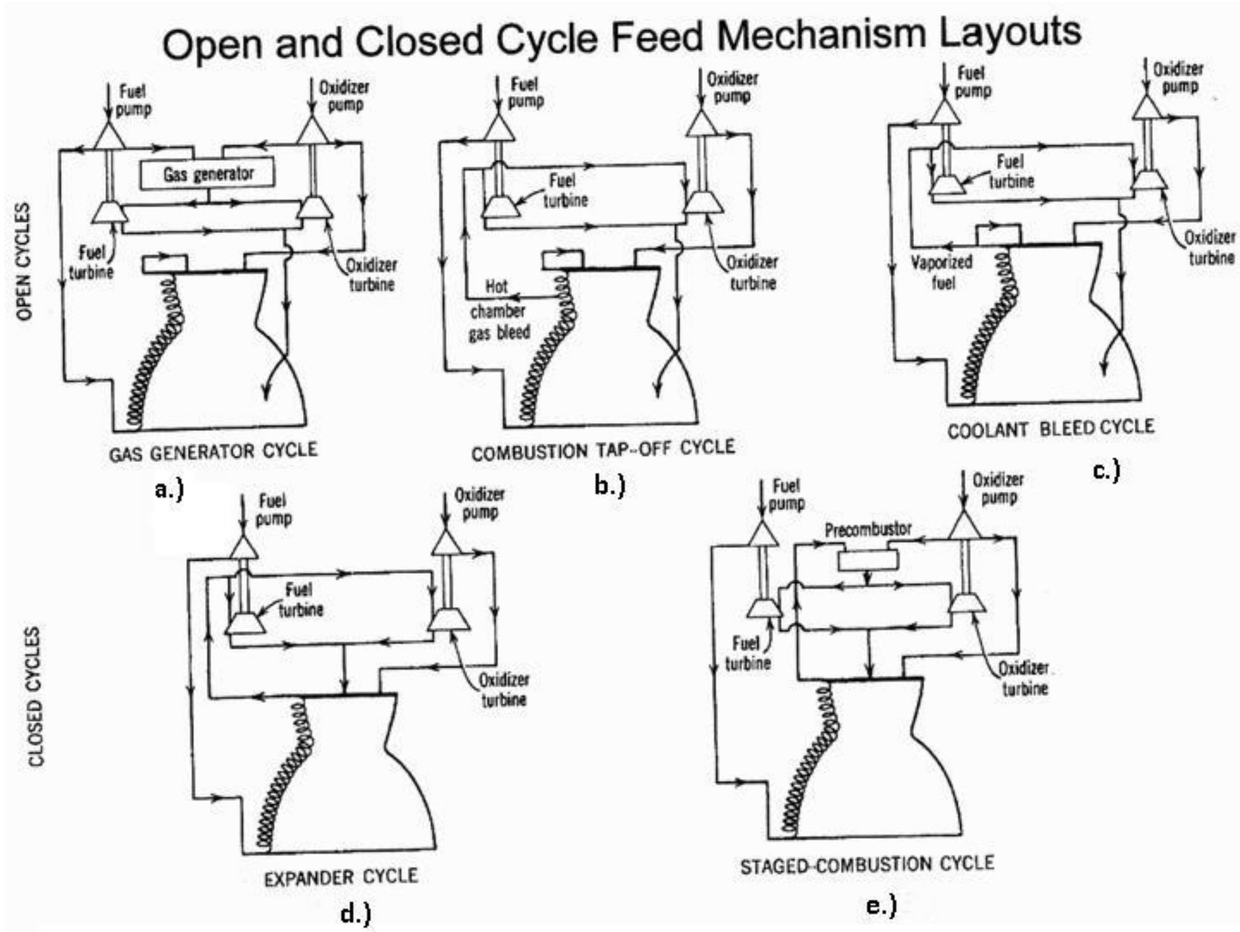

Figure 2: Open and closed cycle feed mechanism layouts

The use of these deep cryogens in the first stage results in much larger, heavier and more complicated tankage and associated structural configurations, plus the need for a heavier and more complex flight thermal insulation system. These additional design needs and difficulties generally result in a relatively poor stage mass fraction, when compared to a much higher density, less volatile, ambient temperature storable fuel such as a hydrocarbon like kerosene (designated as RP-1). Since mass fraction is just as important as I_{sp} in the basic vehicle rocket equation, design solution, a higher density, more storable fuel (at room temperature) such as kerosene is, in many cases, a better choice than hydrogen as a fuel for many booster first stage applications. In addition, the many operational and propellant handling advantages of using room temperature storable kerosene fuel as compared to cryogenic hydrogen, provide even more rationale for selection of the hydrocarbon for some booster stages over hydrogen (as is the case with all Russian launch vehicles, and the U.S. Atlas V and Falcon rockets, as well). Just like LH_2 fueled booster engines, these desirable qualities of hydrocarbon fueled booster engines also have two sub-options for their design approach and implementation. These are also an open (option 3) or closed (option 4) power cycle mechanization.

Other hydrocarbon fuels such as liquid methane, ethane or propane (which are the main components of liquefied natural gas) might also be good choices for higher density, better performance (more hydrogen atoms per molecule than RP-1) booster stage fuels, but to date there is only limited technology experience with these hydrocarbon fuels and liquid oxygen in rocket combustion devices; and at this time there has been no sustained successful flight vehicle experience. Thus, while these higher performance hydrocarbon fuels might become good choices for future advanced launch systems, nearer term rocket engine designs using liquefied natural gas-type fuels are not considered to be at a sufficiently high enough level of technology maturity or readiness for any near term flight application.

However, an example of a newer hydrocarbon fueled engine technology development other than the standard LO_2/RP-1 combination that could be developed for near term flight applications is illustrated by a program recently conducted by the Northrop-Grumman Propulsion and Products Center, under contract to NASA. This effort to develop a smaller than booster class LO_2/methane spacecraft velocity control engine was managed by NASA Glenn Research Center, with technical direction by NASA Johnson Space Center. This combustion device, designated as the N-G/TR-408 and delivering approximately 100LBF of thrust, successfully demonstrated robust operation over widely varying propellant quality conditions.

The TR408 development effort presented some major challenges, including the fact that there was little information about how to build a rocket engine that was capable of burning this propellant combination, especially in both gaseous and liquid states. NASA challenged N-G engineers to come up with an engine that could intake any propellant quality between 100 percent gas or 100 percent liquid without affecting performance or stability. A heat exchanger chamber that vaporized both propellants was the critical design element of the TR408 technology. It was necessary to design an engine that could operate over the full range of density, from gas to liquid. With some clever thermal design work and engineering, the resulting engine design concept demonstrated that it could generate high performance by consuming 5 to 10 percent less propellant mass than conventional hydrocarbon fuel engine designs.

In the course of this technology effort, N-G engineers developed a one pound-force igniter; created a gas-gas injector; linked the heat exchanger chamber with the injector and igniter; coupled the igniter to the main engine's fire valves; and demonstrated the basic operation of this design. Then the N-G propulsion team completed hot-fire demonstration testing, thus validating the prototype design enhancements with a fully expanded nozzle skirt. Propulsion systems using engines like the TR408 have the potential to use LO_2/LCH_4 type propellants produced indigenously from lunar or Martian soil. Theoretically, astronauts could safely travel to Mars to manufacture liquid oxygen or liquid methane as return propellants. Since it is now believed that there is water ice in the Martian soil, and the Martian atmosphere is about 95 percent carbon dioxide, it would then be possible to make an oxidizer which is liquid oxygen, and fuel, which is methane. That would be as safe as the natural gas used in cars or in a stove at home.

3.0 LRHCBE Design Approaches

As described above, selection of LO_2/RP-1 for a booster MPS, as in the case of LO_2/LH_2 MPS engines also has two sub options. These are again, either an open or a closed power cycle engine. Open cycle gas generator engines, as illustrated in Figure 1-b, have been used for a number of booster stages in the U.S., including the Vanguard, Jupiter, Juno, Redstone, Delta II, Atlas 1 and 2, as well as some newer commercial launchers (Falcon 1 and 9) and the Saturn V launch vehicles. The closed cycle hydrocarbon fuel design, which is commonly used in Russia, is designated as an Oxidizer Rich Staged Combustion (ORSC) engine. The U.S. has used several Russian ORSC engine designs for booster stages, but as of yet has not designed and developed a new domestic ORSC engine for any current flight applications.

The ORSC hydrocarbon fueled rocket engine is generally the highest performance design approach possible for LO_2/RP-1 propellants for several fundamental reasons. These include: 1)ORSC engines generally run at relatively high chamber pressure (typically > 3000 psi) which, enables a higher nozzle area expansion ratio, without concern for separation at sea level (lift off condition); 2) operation at higher chamber pressure (P_c) results in higher net I_{sp} than open cycle engines, because of the combined benefits of the higher p_c that slightly improves combustion efficiency and enables a higher nozzle expansion ratio (higher sea level thrust coefficient, or C_F at sea level). A small net I_{sp} gain also is realized, because all of the propellant is fed into the MCC at an optimum mixture ratio (O/F) including the pump drive gas, unlike the open gg cycle, where the small amount of fuel rich pump drive gas is dumped overboard and makes almost no contribution to net thrust; and 3) higher chamber pressures yields more compact, smaller engine sizes and therefore a higher thrust-to-weight ratio than a lower pressure gg powered engine. The characteristics of open cycle gas generator engines (or the "tap-off" cycle, which is an alternative to gg's, and takes some of the hot gas directly from the fuel rich wall zone of the MCC to drive the pumps instead of a separate gg and then dumps the exhaust gas overboard like the gg open cycle- this approach eliminates one combustion device, but complicates engine design and operation with no real net performance gain) versus ORSC closed-cycle engines are compared in Table 1. Simple schematic diagrams of all liquid booster engine options are shown in Figure 2, while the operating characteristics for each engine cycle is summarized in Table 1.

Table 1: LO_2/RP-1 Typical Range of Operating Conditions

Cycle	Nominal Chamber Pressure Range (psia)	Vacuum I_{sp} Range (s)	Thrust/Weight Ratio
Open, gas generator	500 – 1 000	300 – 315	70 – 80
Closed, oxygen-rich staged	>2 000 – 3 500[a]	325 – 350	100 – 120

[a] More typical for Russian engines such as RD-170 and NK-33.

Table 2 summarizes the advantages of an ORSC engine over an open cycle gas generator engine (or tap-off cycle) as well as listing the primary issues/concerns. While the turbine drive gas from the other staged combustion cycle, which uses a fuel rich pre-burner, such as was used for the Space Shuttle Main Engine (SSME) presents a relatively benign condition for all the materials in the hot-gas flow path, it forces the turbine to run at higher temperatures to achieve the required drive power and therefore induces greater thermal stresses and associated life concerns for the engine turbomachinery (especially the turbines and bearings).

A typical ORSC engine is depicted schematically in the very simple diagram in Figure 3. Liquid oxygen and the hydrocarbon fuel are fed into a high pressure preburner, which initially combusts at greater than stoichiometric mixture ratios (MR = 10-20), and then the initial combustion products are flooded downstream in the preburner chamber with large amounts of LO_2. The LO_2 is usually injected in the preburner chamber through some type of tangential slots, such that the effluent gas leaves the preburner at an MR of 60-80 or so and an exhaust temperature of 700-800° F.

TABLE 2: Advantages of Oxidizer-Rich, Staged-Combustion (ORSC) Rocket Engines over Open-Cycle Gas-Generator LO_2/RP Engines

Advantages	**Issues/Concerns**
Higher I_{sp} (up to 7-10%)	Because of its oxidizer-rich hot gas environment, engine components and plumbing (ducts) need to be made from compatible and flame-resistant materials or require the use of special nonburning, resilient protective coatings that do not erode or chip away during handling, testing, and operation (especially important for reusable engines).
Use of higher density fuel enables higher overall mass fraction and more favorable aerodynamics profile rocket stages/vehicle design.	Greater tendency for combustion instability because of the more difficult to burn hydrocarbon fuel operating at much higher chamber pressure in the preburner and main combustion chamber
Because of higher chamber pressures, the engine design results in nozzles with higher sea-level area ratios and significantly higher engine thrust/weight ratios.	Preburner design and development is more difficult than fuel rich gas generators (GGs). Because of high operating pressures, there will be a greater tendency for combustion instabilities and the need for high-temperature oxidizer and flame-resistant materials in the turbines and any associated hot gas ducts and the oxidizer side of the main combustion chamber (MCC) manifolds and injectors.
Results in oxygen-rich shutdown, which minimizes carbon deposits and 'coking" of injector orifices with hydrocarbon fuels – therefore, easier to restart multiple times.	Because of the high preburner operating pressures (6,000-9,000 psia), ORSC engines will require boost pumps and boost pump devices. Fuel-rich GGs typically run at much lower pressures- ~1,000-1,500 psia – and are easier to design with fewer components.

Enables pumps to generate required power at much lower operating temperatures than with fuel-rich GG powered turbines, which results in increased life and durability (typically about 700°F versus ~1700°F for fuel-rich GG cycle).	
Eliminates open-cycle GG exhaust plume interactions and interference issues by running all of the turbine drive gas back into the MCC.	
Usually allows increased engine service life because of generally lower operating temperatures.	

The preburner exhaust gases are then run through the turbopump assembly (TPA) turbine, which typically drives both the fuel and the oxidizer pumps using a gearbox and separation seals. Note that all staged combustion engines must operate with pump discharge pressures significantly higher than the main combustion chamber (MCC) pressure because the main drive turbine operates in series with the MCC. The LO_2 is fed back to the preburner and the oxygen-rich gas from the turbine exhaust is then fed into the MCC injectors, where it is mixed with the liquid RP-1 coming from the fuel pump. Both the hot oxidizer-rich gas and the RP-1 enter the MCC through the chamber injectors. Typically, the injector has hot gas/liquid RP-1 swirl elements. The number of these swirl injector elements depends on the engine thrust level and scales somewhat with engine size and thrust level.

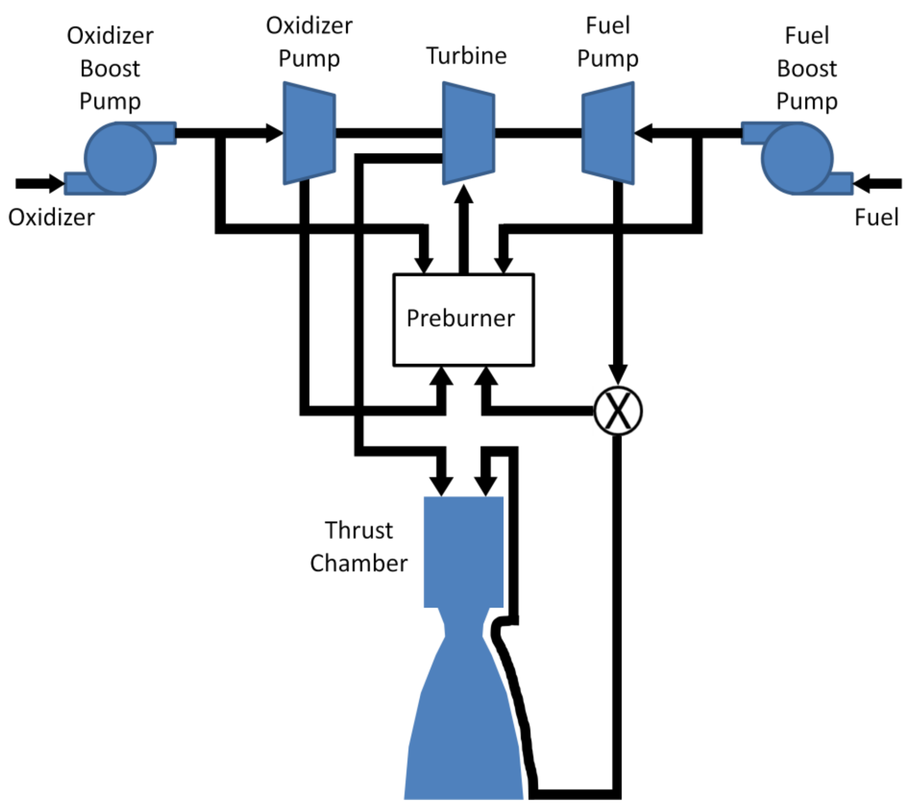

Figure 3: Simplified schematic of a closed-cycle, staged-combustion rocket engine. Because the preburner, turbine, and thrust chamber operate in series, the required pump pressure is higher compared to open-cycle engines which often require the use of a high pressure boost pump. SOURCE: Air Force Research Laboratory

Boost pumps are often used to feed the high-pressure LO_2 and RP-1 into the preburner. The engine start and shutdown sequences and methodology vary and tend to be somewhat complex, but they are usually established by the engine timing, calibration, the internal power balance, operation of the flow control valves, and other fluidic elements. The oxidizer and the fuel pumps are usually mounted on the same common shaft, and dynamic seals and intercavity inert gas purges keep the two liquids well separated. An upstream start turbine is sometimes used to start TPA full operation. The various types of engine cycle designs are summarized and compared schematically in Figures 1 and 2. They are also compared in a little more detail in tables 3 and 4.

TABLE 3: Top Level Liquid Engine Cycle Comparison

	Staged Combustion Closed Cycle Pump Fed	**Gas Generator Open Cycle Pump Fed**	**Pressure Fed No Pumps**
Vehicle Weight	Lighter tanks	Lighter tanks	Heavier tanks
Chamber Pressure	2000-3500 psia	700-1500 psia	300 psia
I_{sp}, Vacuum, LH_2 Fuel	438 sec	405-416 sec	381 sec
I_{sp}, Vacuum, H.C. Fuel	345 sec	310-320 sec	300-310 sec
Nozzle Arco Ratio	78-100	15-24	7
Tank Wall Thickness	Thinner	Thinner	Thicker
Tank Pressurization System Required	Yes, Smaller Low Pressure System	Yes, Smaller Low Pressure System	Yes, Large High Pressure System
Tank Pressure	30-40 psia for pump NPSH	30-40 psia for pump NPSH	500-600 psia Total Feed System Forcing Pressure
Vehicle Complexity	Very High	High	Low, Simple
Approximate Engine Component Parts Count	3047 – 5807*	400 – 600	80-100
Booster Engine Cost	Most Expensive	Less Expensive	Very Cheap BDB Concept – but very limited throw weight, low payload to vehicle weight ratio (glow)
Manufacturing Complexity	Very High	Medium	Very Low

*based on the U.S. SSME

TABLE 4: Description of Liquid Rocket Engine (LRE) Cycle Options

Open or Closed Cycle Feed Mechanisms

Open Cycle – Turbine exhaust is discharged at low pressure point into engine nozzle or out of one or two separate low pressure exhaust nozzles

Closed Cycle – Turbine exhaust is injected into combustion chamber

- Higher I_{sp} (5-10%) because turbine exhaust goes through full pressure ratio of engine, contributing directly to full thrust
- Pump turbine must operate at a higher pressure than an open cycle turbo-pump

Two types of closed cycle LREs

- Expander – Is only used thus far with LH_2 fuel i.e., RL-10 for the centaur upper stage
- Staged Combustion – Fuel rich, SSME only oxidizer, rich, all Russian engines are (ORSC) and is increasingly in use elsewhere in the world
- Full flow staged combustion, FFSC never developed, but demonstrated at subassembly level with the USAF-IPD program

Open (drive gases do not go through throat)

Gas Generator

- Some propellant is diverted into a smaller chamber to generate drive gases
- Example: F-1, J-2

Tap-off cycle

- Some gas is bled directly from the combustion chamber to drive turbines
- Example: J2-S

As a general rule, open cycles are slightly lower performance (5%-10% lower I_{sp}) than closed cycles

- Staged Combustion, (FRSC) SSME (fuel rich) in USA oxidizer rich, all Russian LREs
- Expander Cycle, RL-10 for Centaur upper stage

4.0 LRE Engine Control

Some type of engine controller is always necessary to ensure full capability (including mixture ratio control) and reliability for a modern LO_2/Hydrocarbon engine. Today's engine controllers are typically of the very reliable, full function capability digital electronics design. The most important engine operating functions that the controllers are used for, are engine throttling during ascent (to compensate for the extreme atmosphere stresses that occur at "Max Q") and mixture ratio control over the entire boost phase to ensure that all the propellant is used at the design flow rates for both fuel and oxidizer and that the propellant residuals at the end of boost are as near zero mass as possible. The controller can also be used for redundancy and reliability management and for integrated vehicle health management to the degree that these functions may be designed into and used by the launch vehicle.

Engine throttling, which is usually implemented together with mixture ratio control, is accomplished with a complex combination of flow bypass valves, and fixed bias and calibration orifices that are inserted during engine and hydraulics system build-up, calibration, and hot fire tests. All liquid RP-1 enters the thrust chamber cooling jacket prior to entering the MCC injectors and is mixed with the oxidizer-rich preburner exhaust gases to achieve the final main chamber combustion process. The liquid cooling jacket (heat exchanger), together with some film cooling of the MCC chamber wall, maintains the MCC at an acceptable operating temperature while providing the necessary engine combustion efficiency and associated I_{sp}.

5.0 Design Approach and criteria that must be established for, as well as Potential ORSC Risks that may need to be addressed During a New Engine Development Program

It is believed that development of a new American ORSC engine design could involve a number of potential risk areas that would need detailed analyses. test, evaluation, and successful resolution prior to full flight qualification. These potential risk areas include:

- *Combustion stability.* Combustion stability physics for high-pressure liquid-liquid preburners and the MCC gas-liquid injectors LO_2/RP-1 ORSC engines are not well understood in the U.S. rocket industry. Combustion instability issues have plagued many hydrocarbon fuel rocket engine development programs during the past 60 years. Both physics-based modeling and a well-defined test program will be required to achieve and demonstrate the required stability margins for both combustion devices and thereby retire these risks. In the U.S., the Office of Air Force Scientific Research, in close cooperation with the US AFRL's rocket propulsion laboratory at Edwards Air Force Base, has been making significant investments to analyze, predict, and defeat the combustion instability problem in liquid oxygen, hydrocarbon (LO_2/HC) engines.
- *Injectors.* The new ORSC injector will most likely be based on some type of a co-axial swirl design that will have to be tuned to MCC frequencies and set up with either acoustic cavities and or some type of baffles to ensure stable engine operation. Also, thrust-scaling relationships will have to be established through a combination of analysis and empirical data. While this is not presently seen as a major risk, there will have to be a dedicated experimental testing effort to tune the injector and chamber at the subscale and then the full-scale levels, depending on the required thrust level.
- *Operation in high-pressure, oxygen-rich environments.* High-pressure, oxygen-rich environments can be very hard on inert materials, and dangerous conditions can result following component failures that are difficult to contain when trying to recover to a safe operating state, or at engine shutdown. This type of environment is unique to the ORSC engine. As described above, in an ORSC engine all oxidizer consumed in the engine combustion process is first used to drive the TPA turbine, resulting in high-power margin and a relatively low operating temperature (as compared to fuel rich staged combustion). The resulting oxygen-rich environment is also relatively clean, such that no soot or other residual combustion product deposits are generated during normal operation. Because of these unique conditions, the ORSC cycle generates high-

pressure oxygen rich environments that present compatibility challenges for traditional turbine and hot gas ducting materials.

- *Physics-based analytical models.* Another risk for the RBS MPS is a fundamental lack of fully anchored physics-based analytical models in the U.S. for ORSC engines. In the anticipated fiscal environment of limited budgets and tight and optimistic development schedules currently available for any new technology development, reliable and accurate analytical models and tools are critical for the successful and cost-effective completion of any new engine development DDT&E program. Accurately anchored models enable a reduction in the number of design and fabrication cycles and, most significantly, expensive test cycles that were required for past rocket engine development programs. The analytical model development needs to be multiscale, with modeling ranging from subscale components to subassembly levels such as the power head (TPA, preburner, and flow control valves) as well as at the fill-scale ORSC engine level. This approach, carefully and logically applied will ensure that validated models will be available at each step of the design process for a new ORSC engine when scaling to higher thrust levels. These same models will also benefit other future ORSC engine development work by accelerating and reducing the total number of expensive tests required.
- *Valves/sensors/actuators.* Some development effort will be required to obtain the necessary highly reliable advanced fluid flow control valves and other control elements to enable a wide range of throttling, mixture-ratio control for proper and efficient propellant utilization, engine balance and calibration, thrust vector control (TVC), IVHM, and various other engine controls. These control elements must have highly durable, reliable, modern, accurate sensors and should be fully integrated with an automated digital MPS and engine control system, that is, in turn integrated with the reliable vehicle guidance and control system, and with any integral health management system (IVHM) that may be incorporated in the engine design.
- *Systems Engineering.* A firm definition of the mission application needs and/or requirements for system engineering and integration of the MPS and the other major systems for a new launch vehicle must be clearly established. These requirements must be firmly set early in the development process to avoid serious conflicts, unnecessary complications, and, worst of all, requirements creep, before the program critical design review and engine flight certification. If these design requirements are not firmly established and maintained early in the program, there will be a high probability of serious cost overruns and schedule slippage.
- *Power balance.* Overall engine power balance and flow calibration and control with robust margins and tolerances for a wide range of operating variances must be established early and verified by testing. Otherwise there will certainly be problems with off-nominal operating conditions, which often occur later in the program or, worst of all, during actual flight. There must be demonstrated anomaly and off-nominal operating capability and robust margins designed into the more complicated OSRC engine to avoid failures that would be catastrophic during flight.
- *Turbomachinery.* There are always risks with new high speed engine turbomachinery. Because of the high-pressure preburner operation, boost pumps often need to be used in the engine and as part of the overall cycle. Engine start-up transients and shutdown sequences will all have to be established and fully characterized for this more complicated ORSC, multicomponent engine to ensure safe and repeatable operation. Based on past experience this is not considered to be a major concern, but is a moderate risk that will have to be addressed during the design and development process with focused, dedicated analytical and test efforts that will have associated costs.
- *Long-life bearings.* High-speed, long-life multiple start turbomachinery shaft bearings will have to be developed and verified for all ORSC engine rotating machinery. This is a low-to-moderate risk concern but one that must be explicitly addressed, because bad bearing choices and their subsequent integration into the engine can lead to serious problems later in the DDT&E program. This happened more than once in past programs such as the SSME, when bearing issues were discovered after a number of “re-usable” space shuttle flights were perceived to be successful.
- *Materials.* The last risk, a moderate one, is the mechanical and dynamic design approach and especially the materials to be used for oxidizer-rich, hot-gas ducts from the preburner, which may require flexjoints or axial joints and other complications. As mentioned above, this risk area will have to be empirically evaluated before committing to a final engine design.

Thus, the principal technical risks associated with the development of a new LO_2/RP ORSC rocket engine will be associated with combustion stability and operation in the high-pressure oxygen-rich environment. Significant additional basic and applied research on combustion stability will be necessary before analytical tools are available that allow confident and repeatable prediction of combustion instabilities. There is some of this work currently under way in the U.S. at both USAFRL and NASA, but significant improvement in prediction capabilities cannot be anticipated in the near term. Fortunately, empirical techniques are available to "de-tune" the combustion system if instabilities arise, so this risk is principally one that will result in needing additional development time and resources if instabilities arise as the engine is scaled up during its development phase from sub-scale hardware.

The risk associated with the oxygen-rich operation is fundamental and potentially more difficult to overcome. It is well known that Russian engine designs have overcome this material incompatibility challenge by using inert enamel coatings on traditional high-strength turbine alloys and hot-gas ducting. The alloys provide the structural load support, while the enamel coating provides the requisite hot-oxygen-rich environment protection for various exposed surfaces. This type of solution has been used on Russian ORSC rocket engines for over 50 years and is also used around the world for hot gas-turbine applications in jet engines and domestic power generators. There are at least three flight-certified Russian ORSC engine designs (RD-170, RD-180, and NK-33) that are well known to the U.S. rocket engine industry but not produced in the United States. Through various business arrangements with the Russian engine manufacturers, the basic design for these engines is well known and understood by the U.S. manufacturers – namely, Pratt and Whitney Rocketdyne (PWR)[1] and Aerojet.[2] Each company supplies a version of a different Russian engine (e.g., PWR sells the RD-180 and Aerojet sells a modified NK-33) to U.S. launch vehicle suppliers.

Another oxygen-rich compatibility approach is to develop and use a hot-oxygen gas compatible inert parent material.[3] This approach is being investigated by the AFRL and large engine contractors (PWR and Aerojet) with some successful results already having been reported. PWR has developed a hot-oxygen-compatible material called Mondaloy and has been evaluating its applicability and durability in hot oxygen-rich environments. To date, only small coupon samples of Mondaloy have been manufactured, and the statistical basis to establish the thermal and mechanical properties needed for engine design and development is currently lacking. As such, basic issues with weldability, fatigue, and fracture mechanics remain to be investigated and design criteria then firmly established.

Currently, the Russian-developed enamel coatings are a far more mature and proven technology. However, application of these special coatings and/or advanced materials to U.S. ORSC engine designs has not been fully proven, so a comprehensive risk reduction program will be required. This risk mitigation effort must be focused on developing and proving a new hot-oxygen-compatible parent metal alloy or verification of a coated material system capable of multiple reuses for the turbine and hot-gas flow-path components similar to the Russian solutions. Thus, if the known coatings, which are inert and will not react with hot oxygen, become the preferred approach, the risk mitigation effort will have to ensure that the coated engine elements are sufficiently durable under all the necessary environmental exposures. In summary, a new ORSC engine development program will have to verify coating durability under all relevant conditions and demonstrate traditional hot oxygen environment compatibility in order to be certified for a new flight LRHCBE application.

It is believed that in spite of the ORSC engine risks and concerns discussed above, there already exists an extensive database with successful experience with the use of this type of engine around the world. Table 5 lists all currently known LO_2/HC engines using either open-or closed-cycle designs that have been flown or are flight qualified throughout the world that should help to give high confidence in achieving success in developing a new advanced ORSC engine. As can be seen from this table, ORSC engines have been, or are about to be flown on launch vehicles in the United States (all Russian designed and built), Russia, Ukraine, India, China, and South Korea. This extensive successful development and flight history and development experience around the world should provide

[1] A. Weiss, Pratt and Whitney Rocketdyne, "Reusable Hydrocarbon Rocket Engine Maturity for USAF R BS," presentation to the Committee for the Reusable Booster System: Review and Assessment, February 16, 2013, Approved for Public Release.

[2] J.Long, Aerojet, "Reusability and Hydrocarbon Rocket Engines – Relevant US Industry Experience," presentation of the Committee for the Reusable Booster System: Review and Assessment, February 16, 2012. Approved for Public Release.

[3] R. Cohn, Air Force Research Laboratory, "Hydrocarbon Boost Technology for Future Spacelift," presentation to the Committee for the Reusable Booster System: Review and Assessment, February 16, 2012. Distribution A-Approved for Public Release.

confidence for the relatively straightforward development of a new LO_2/HC engine powered booster. Many of these LO_2/HC powered launch vehicles have successfully placed large payloads of all kinds into their required Earth orbits or onto space-science trajectories throughout the solar system. This long history of success with various hydrocarbon fueled engines also includes putting many human beings in space as well as on the moon. So, these past successful experiences with both open and closed cycle LO_2/HC engines certainly provide assurance that a completely new ORSC engine can be developed if adequate resources, time, and planned reserves are devoted to an affordable and reasonable program. Figures 4 and 5 graphically display and summarize the known performance capabilities for some of the world's HC engines that have been used, or are in use today, for booster stages.

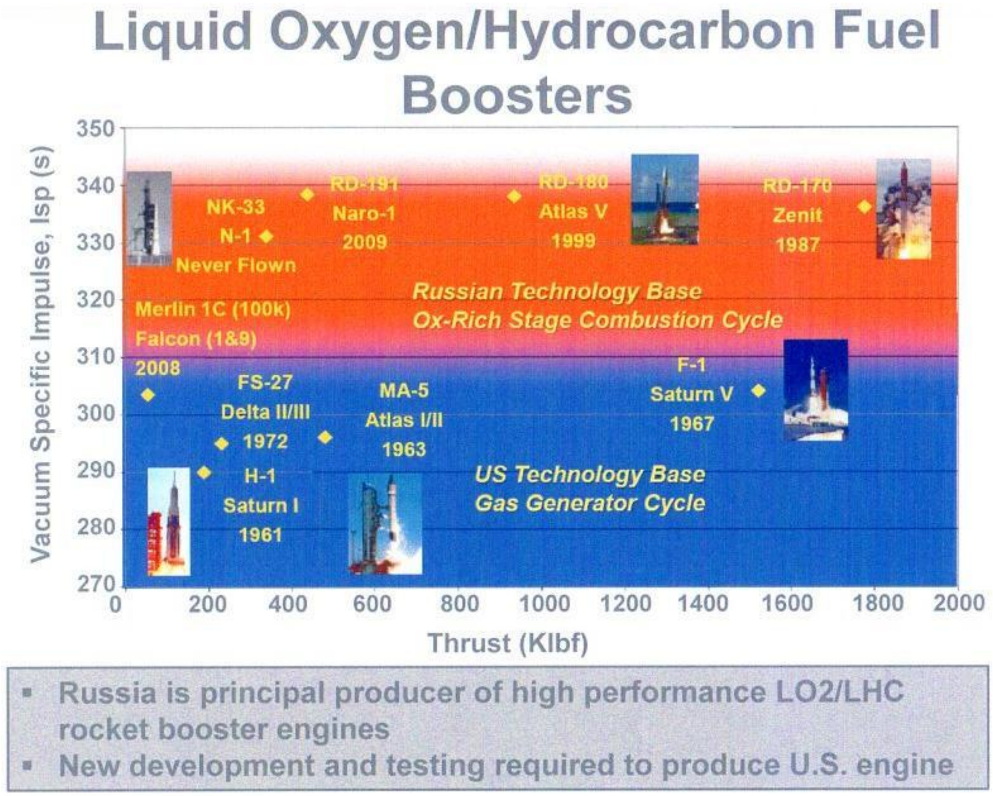

Figure 4: Performance of various liquid oxygen / hydrocarbon booster engines

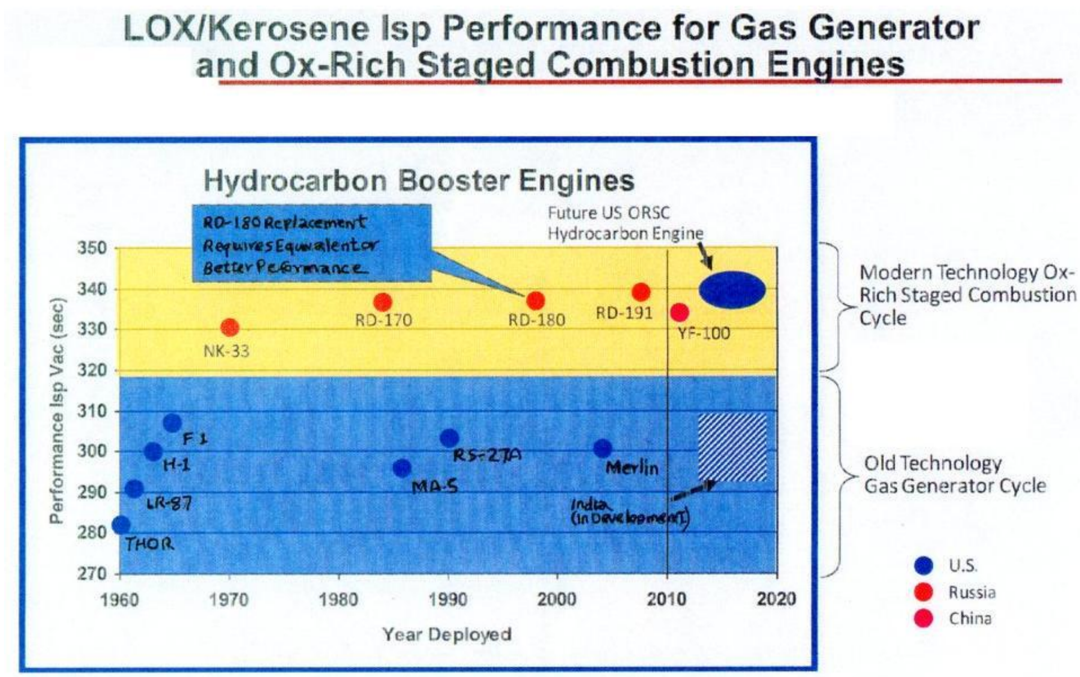

Figure 5: Performance history of various hydrocarbon booster engines

6.0 Government Sponsored Hydrocarbon-Fueled Booster Engine Risk Mitigation Efforts

The U.S. Air Force (AFRL) has recently been pursuing subscale engine hardware technology demonstrations and risk mitigation programs to advance ORSC engine technology. These R&D type programs are briefly described below together with a short overview of NASA's hydrocarbon engine development activities.

The AFRL has been conducting a joint rocket technology program with industry, the Integrated High Performance Rocket Propulsion Technology (IHPRPT), to advance all forms of rocket propulsion technology, including hydrocarbon boosters, for about the last 20 years.

Some critical ORSC technologies are currently being studied and developed under ongoing AFRL-IHPRPT propulsion R&D programs. The hydrocarbon boost (HCB) Phase II demonstration program is intended to develop technologies to support advanced ORSC LO_2/RP-1 engine capability. Conducted by both Aerojet and PWR, this program aims to mature advanced hot-oxygen-rich-compatible materials and coatings as well as engine components such as pumps, advanced hydrostatic bearings, valves, actuators, preburners, igniters, main thrust chambers and new engine controllers, IVHM systems and associated sensors. A potential approach and methodology for on-board automated flight controls and fault management, using the booster engine controller integrated with an IVHM system and the vehicle flight controls is shown in the simplified diagram of Figure 6. This approach is currently being explored by the USAFRL for application to future Air Force launch vehicles.

NASA has also been conducting advanced OSRC engine development programs over the last 15 years, working to solve many of the same advanced technology problems as those that the Air Force has been addressing.[4] Several years ago NASA Marshall Space Flight Center (MSFC) funded a new ORSC engine program known as the RS-84.

[4] G. Lyles, "NASA's Reusable Stages and Liquid Oxygen/Hydrocarbon (LOX/HC) Engines," presentation to an NRC Committee for the Reusable Booster System: Review and Assessment, February 17, 2012. Approved for Public Release.

This engine program proceeded through the preliminary design phase and was able to conduct some advanced prototype component design, manufacture, and test before it was canceled because there was no formal mission requirement and the NASA budget did not provide sufficient advanced technology funds. Nevertheless, some successful component-level results were achieved that are now directly applicable to a new ORSC engine development program, if one were initiated.

NASA also worked jointly in the past with the Air Force on a relevant advanced technology program known as the Integrated Powerhead Demonstration (IPD). The IPD program was initiated by the Air Force but soon evolved into an effort jointly funded by AFRL and NASA/MSFC with all hot-fire testing conducted at the NASA Stennis Space Center. The power head was to be integrated into a full-flow staged-combustion engine, where all propellants (in this case either LO_2 / LH_2 or LO_2 / RP-1) flow through oxidizer-rich and fuel rich preburners that power separate fuel-pump and oxidizer-pump turbines, respectively. The hot-gas exhausts from each preburner flows into the MCC through a gas-gas injection system, making 100 percent of the propellant energy available to produce thrust. The IPD was designed as a ground-based demonstrator for an engine that would produce 250,000 lb. thrust. It was designed, built, and successfully tested, thereby demonstrating compatibility for high-performance and long-life components, materials, and technologies for new booster engine applications and (for the first time) a gas-gas injection MCC. After the successful tests at NASA Stennis, the program was terminated by the government, also for a lack of funding and a lack of a well-defined mission requirement. The successful IPD demonstration led to an improved understanding of advanced engine components and many of these results with the hot oxygen rich gas side of the system will be directly applicable to advanced component development associated with any type of new ORSC engine.

NASA is also exploring the development of a new ORSC engine in the million-pound thrust range for its Space Launch System (SLS) Advanced Liquid Strap-on Booster (ALSB). There are conceptual design activities that are on-going for a program that could demonstrate significant risk mitigation results for a new ORSC engine.

NASA has also announced plans to develop advanced technology materials and components that are applicable to both the SLS ALSB program and any new launcher ORSC program.

TABLE 5: American Heritage or Other Available Source Hydrocarbon Rocket Engines: Previously or Currently in Use, or in Development

Rocket Engine	**Manufacturer/ Supplier**	**Cycle**	**Thrust Level (lbf)/Vacuum Specific Impulse (s)**	**Status**	**Application**	**Comment**
RS-27 MD-1 MA-7	Pratt and Whitney Rocketdyne	GG	200K S.L., 237K ALT/303	Flown hundreds of flights (~800)	Delta II and A2L Previous Thors, Thor Delta, and Delta III	
MA-5A LR-89/ LR-105	Pratt and Whitney Rocketdyne	GG	430K (booster) +60 (sustained) / 297	Flown hundreds of times on Atlases; Project Mercury many flights (1,404)	Atlas family up to satellite launch vehicles	
H-1	Pratt and Whitney Rocketdyne	GG	200K, S.L., 205K ALT / 301	Flown many times (152)	Saturn 1B, Jupiter, and early Thor Deltas	

F-1	Pratt and Whitney Rocketdyne	GG	1,522K S.L. / 307	~65 flights	Saturn V/Apollo	5 used in first stage
RD-180	Pratt and Whitney Rocketdyne, Russian-derived NPO Energomash	ORSC	~860K S.L., 933.4K ALT / 337	~10 flights	Atlas III and V	Two TCAs, one pump
RD-170	NPO Energomash	ORSC	~1700K / 337	Flown many times	Buran, Zenit, Proton	Four TCAs
S-3D	Pratt and Whitney Rocketdyne	GG	80K / 310	46 flights	Jupiter /Juno	
AJ-87 AJ-1 AJ-3	Aerojet	GG	300K / 249	Flown many times on ICBM test flight	Titan-I second stage	
AJ-91	Aerojet	GG	80K / 310	Flown many times on ICBM test flight	Titan-I second stage	
NK-33 (AJ-26)	Aerojet	ORSC	340K S.L., 380K ALT / 330	Intended for use on Russian N-1 Moon launcher first stage	Developed for Russian N1, to be used on Taurus II (Antares)	PC = 2,109 psia
NK-39	Khrunichev/Aerojet	ORSC		N-1 second stage	Developed for Russian N1; intended for use on K 1	
RS-84	Pratt and Whitney Rocketdyne	ORSC	1,050K S.L., 1,123K ALT/338	Finished PDR; cancelled by NASA	Intended for 100 missions, reusable launch vehicle	Incorporates advanced technology items, advanced materials, enhanced water-cooled nozzle
Merlin family	Space-X	GG	~80K / ~302	Flown	Falcon I, 9 and 27	Privately funded development
Other miscellaneous Russian	NPO Energomash	ORSC	Various / ~330	Some flown	Various Russian and Ukrainian Rockets	RD-180
MC-1 (Fastrak)	NASA/Marshall Space Flight	GG	~75K / ~280	Advanced development, many tests	None to date, almost for X-34	
YF-100	China, Inc	ORSC	260K S.L., 301 ALT /	Flown on Long March	China's Long	

	336	March launch vehicle 5, 6, and 7

Automated Guidance and Controls (AG&C) for Automatic Fault Detection and In-Flight Real Time Correction: For Expendable and Re-usable Launch Vehicles (ELV and RLV)

In general, an AG & C algorithm consists of three components: Trajectory Generation, Adaptive Guidance, and Reconfigurable MPS and Control. A diagram of the AG&C components and their interactions with the vehicle's Onboard Diagnostics and Operations Control Center are shown below.

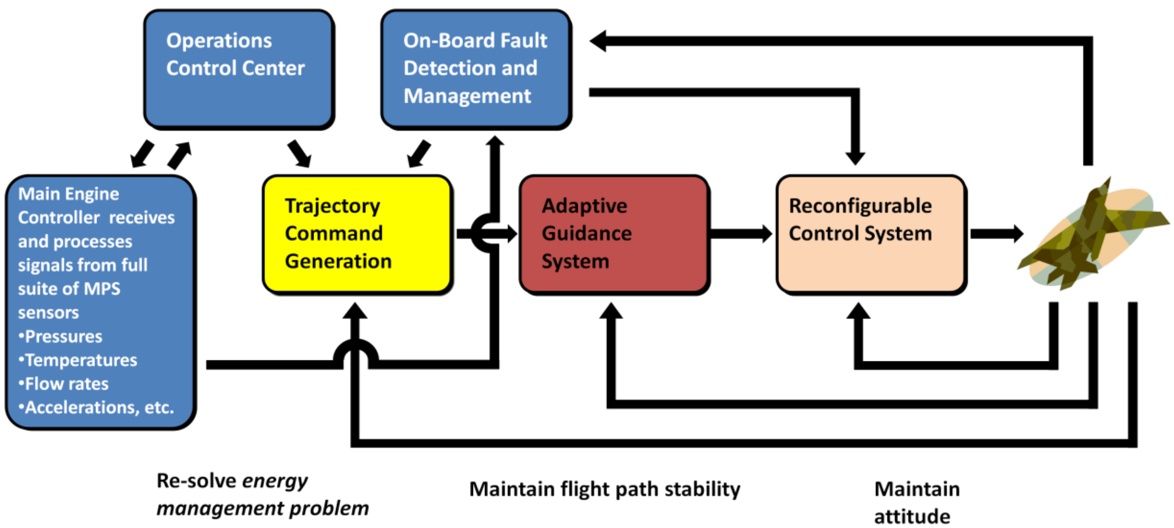

Figure 6. Schematic of the Major Elements of a Fully Automated/Adaptive Guidance and Control System

In the Reconfigurable Control System block, an allocation algorithm distributes stabilizing and compensation commands to the flight controls (such as TVC actuators, engine throttling or other thrust compensation, RCS thrusters, etc.). The stabilizing commands are computed according to the compensation forces and moments required to maintain the stability of the vehicle. In the event of an off-nominal condition, such as a low thrust engine or controls actuator frozen in place, the Onboard Diagnostics detects and provides the failure information to the Reconfigurable Control System. Vehicle constraints and or limits on angular rates, accelerations, loads, and acceleration/velocities are considered by the Reconfigurable Control System algorithm.

In the Adaptive Guidance System block, the vehicle's flight path stability is maintained under both nominal and off-nominal conditions. Feedback gains are adjusted according to the inner-loop to control performance as well as the vehicle dynamics.

In the Trajectory Command Generation block, a new trajectory for the vehicle is generated when necessary so that the trajectory requires only forces and moments that can be achieved by the degraded capabilities of the vehicle while meeting critical constraints.

As the risk associated with hydrocarbon booster engine development is mitigated, the performance of a new U.S. rocket engine will eventually be demonstrated through extensive testing. Previously, new engine verification required years of very expensive testing. These efforts were being performed before very sophisticated computer design and simulation tools had become available. It is now believed, as described earlier that as computer capabilities increase, the traditional testing once required can be significantly reduced such that "cut-and-try" methodology will be replaced with “simulate, test, verify and improve.” However, the cost savings associated with this new approach has not yet been fully demonstrated.

In conventional engine development, new engine design validation testing includes testing at the subscale, component, and subsystem level as well as in full flight configuration. At the component level, the required test facilities can be modest and typically include fluid flow and high-pressure testing of injectors, nozzles, pumps, and thrust chambers. These modest facilities can be used to conduct injector spray tests with cold-flow tests of the regenerative part of the combustion chamber before the full testing of ignition and ramp-up tests over a wide range of operating conditions. Engine combustion stability through all throttle levels as well as dynamic and spontaneous stability conditions is a key performance metric for both the MCC and the high pressure preburner. Sea-level and altitude simulated testing can be performed depending on the application. In addition to reviewing the engineering data after each subsystem test, the components are evaluated for potential failure. This testing is generally instrumental in allowing verification of overall performance over a wide range of performance conditions as well as to validate or provide feedback to analytical models. If sufficiently engineered into the test plan, early testing and evaluation can be used to improve system design for better performance, reliability, and safety. Detailed verification testing typically uses instrumentation, including flow meters, steady-state pressure transducers, thermocouples, high-frequency pressure measurements, strain gauges, accelerometers, and sophisticated laser and optics techniques[5] to provide detailed information on the flow fields and performance of large rocket engines under widely varying conditions.

As the required testing moves toward full system-level testing, PWR, and Aerojet will use the NASA Stennis facility, a national test facility allowing a range of rocket propulsion testing from component-to engine- to stage-level testing. There are also limited rocket test facilities at AFRL/Edwards Air Force Base, commercial facilities at the Mojave Space Center, and contract facilities such as Wylie Labs. These facilities could be available as backup or to handle overflow and surge needs. Additionally, Space-X has developed a facility for their own use at McGregor, Texas.

6.0 Summary of Needed Advancements in Rocket Propulsion Technologies, that should be Sponsored and Supported by the U.S. Government for Future U.S. Launch Vehicle and Space Systems.

There have been very few new liquid rocket engines (LREs) or solid rocket motors developed into flight applications in the US in the last five decades. In fact, in the large booster LRE class during this long time period, only two new large rocket engine development programs, sponsored by the U.S. government were completed and integrated into real flight applications. These were the Space Shuttle Main Engine (SSME) and the RS-68 engine now being flown on the Boeing Delta IV Launch Vehicle. Both of these engines were powered by LO_2 and liquid hydrogen propellants. In addition, two new booster and upper stage LREs were developed with private commercial funds. These were the Merlin first stage and the Kestrel second stage LREs developed on company funds by Space Exploration Corporation (Space-X) and are now flying on the FALCON Family of Space launch vehicles (Falcon 1, 9 and soon on the Falcon 27 or “heavy”). These LREs run on LO_2 and kerosene (RP-1) fuel. A test firing of a Space-X Merlin engine is shown in Figure 9. Two other hydrocarbon fueled (with LO_2 oxidizer) LREs (both ORSC cycle) were imported from Russia and are now being used for U.S. Space launch flight programs. These are the RD-180 flying on Lockheed Martin’s Atlas V and the NK-33, recently modified by Aerojet and then renamed the AJ-26. The AJ-26 Russian design based ORSC LRE has flown on OSC’s Antares launch vehicle for NASA.

A relatively large (about 25) number of U.S. government sponsored LRE development programs were initiated by both NASA and the USAF, but all of these, after a considerable expenditure of funds, were cancelled before any of them ever reached any serious level of prototype design, fabrication and/or testing because the required program funding level was unsustainable and the interest and or need slowly disappeared. The majority of these cancelled LRE unfinished development programs are summarized in Table 6. In addition, the history of all U.S. flight LREs developed and flown over the last 65 years (up until 2010), are summarized in Table 8.

[5] NASA, Rocket Engine Technology Test Bed Practice, NASA Preferred Reliability Practices, Practice No. PT-TE-1427, available at http://engineer.jpl.nasa.gov/practices.html, p.4.

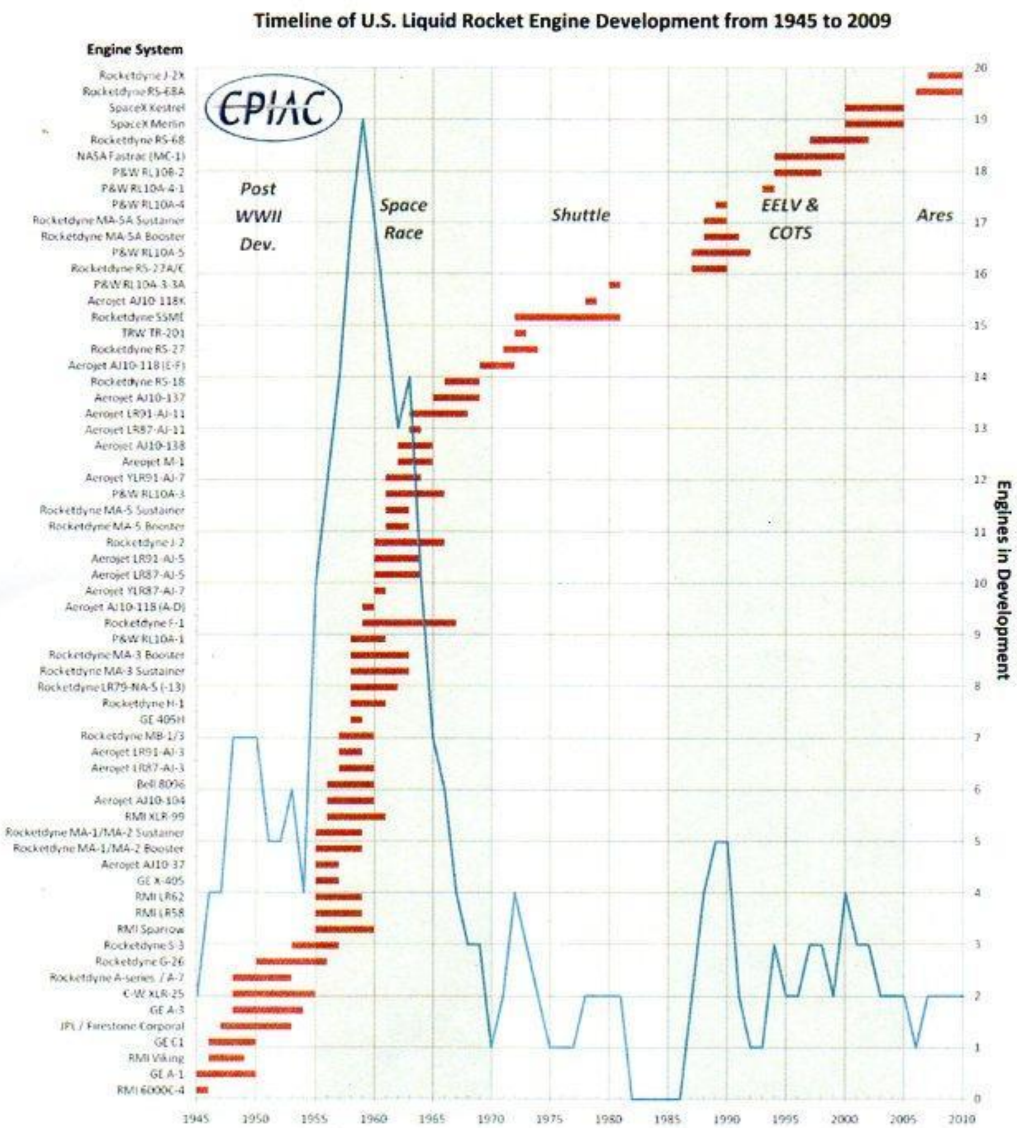

Figure 7: Background and history of U.S. liquid rocket engines developed and/or flown from 1945-2010

7.0 LHRES for Emerging Commercial Launch Vehicles in the U.S.

The emerging commercial space entrepreneurs appear to have been relying upon flight proven but older propulsion technologies that were recognized early on as being low cost designs for their respective launch vehicle stages. Obviously, this approach results in lower costs and risks, but not necessarily higher performance. For example, the engines that power booster and upper stages of the Space-X Falcon family of engines are based on the same original 1960's engine technology that was used on the Apollo program for the landing of the Lunar Module on the moon's surface six times. This engine design approach utilizes a single element co-axial or "pintle" injector as illustrated in Figure 8. This approach is believed to have been selected by Space-X because of its inherent stability and simplicity to fabricate and assemble. Similarly, the Dragon capsule that was the first commercially designed, developed and manufactured

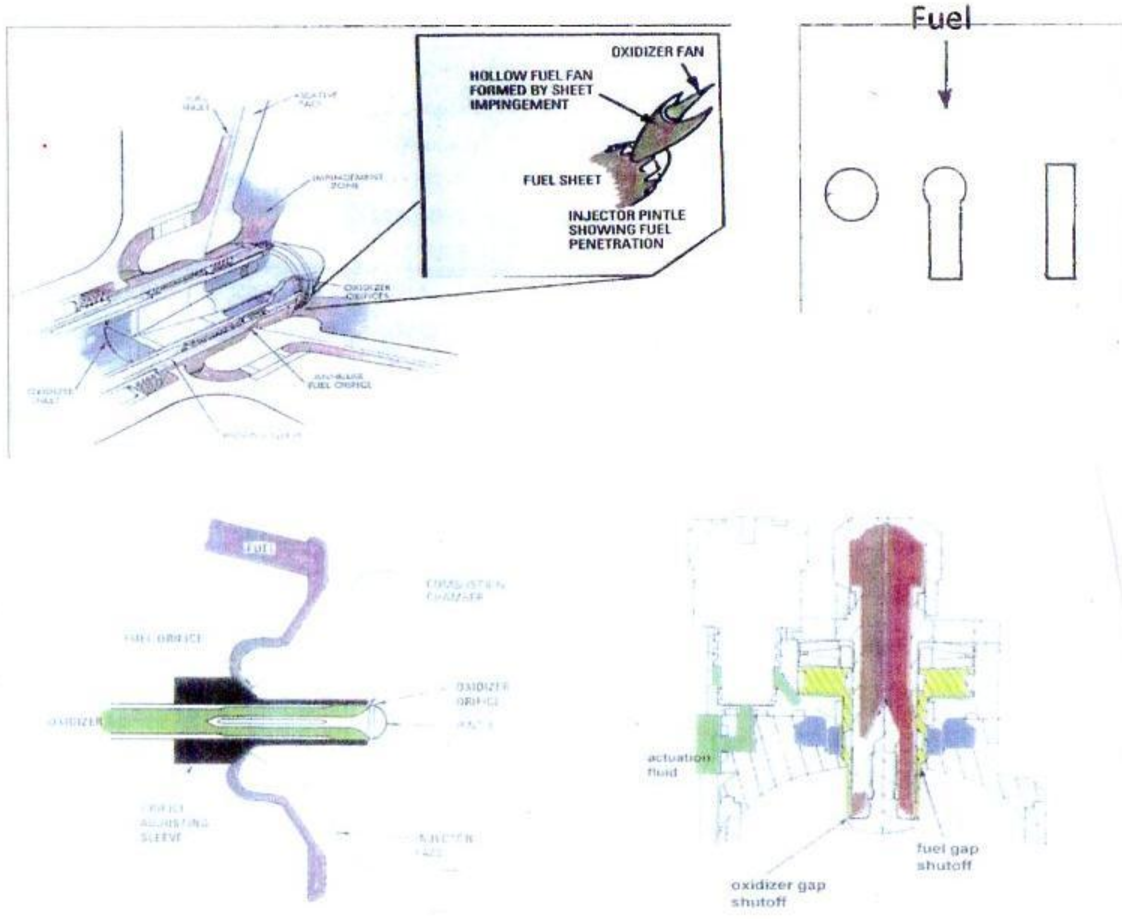

Figure 8: Basic schematic diagram of single element coaxial rocket engine injector also known as "pintle" type liquid propellant injector. Generates cylindrical sheet on radial spray fan. Fixed thrust, throttling, and face shut-off mechanizations. Flown multiple times; used for Apollo Lunar Module descent engine N_2O_4/Aerozene-50, Delta second stage engine N_2O_4/Aerozene-50, numerous space craft apogee insertion engines N_2O_4/MMH and N_2O_4/hydrazine, Falcon I Merlin and Kestrel engines, Falcon 9 Merlin 1st and 2nd stage engines, other commercial launch vehicle booster and upper stage engines in development, and all launch vehicle engines using LO_2/RP-1

Table 6: Cancelled propulsion programs

				Program Maturity								
Engine Program	**Application**	**Company**	**Customer**	**Program Maturity**	**Component Development**	**Demonstration Testing**	**Protoqual**	**Qualification**	**Full-Scale Development**	**Flight**	**Production**	**Costs $**
Transtar	Upper stage	Aerojet	A/F									
Uprated OME	Shuttle	Aerojet	NASA									
XLR-132	High-performance upper stage	Rocketdyne, P&W, and Aerojet	A/F									
XLR-134	Cryogenic upper stage	Aerojet	A/F									
STME	Low-cost booster engine for NLS	Rocketdyne, P&W, and Aerojet	NASA									
STBE	Low-cost booster engine for ALS	Rocketdyne, P&W, and Aerojet	NASA									
LOCUS	Low-cost upper stage	Aerojet	A/F									
Agena-2000	Low-cost upper stage	ARC and Aerojet	A/F									
X-33 RCS	Low-cost RCS engines	Aerojet	NASA									
Cobra	RLV booster engine	P&W and Aerojet	NASA									
ARRE	Advanced peroxide upper stage	Aerojet	A/F									
RS-83	RLV booster engine	Rocketdyne	NASA									
FFSC	Reduced to IPD	Rocketdyne	A/F NASA									
RBCC	Advanced technology	Rocketdyne, P&W, and Aerojet	A/F NASA									
TBCC	Advanced technology	G.E.	A/F NASA									
RS 2200	X-33 linear aerospike	Rocketdyne NASA										
Fastrac	Whatever works (Bantam, X-34)	MSFC	NASA									
SSME, Block III	Shuttle	Rocketdyne	NASA									
LO_2/CH_4 main engine	CEV	TBD	NASA	Cancelled								
LO_2/CH_4 RCS	CEV	TBD	NASA	Cancelled								
RS-84	ORSC for new booster engine	P&W, Rocketdyne	NASA									
HBE	ORSC engine & supporting modeling	Rocketdyne, P&W, and Aerojet	USAF RL									
USET	LO_2/LH_2 upper stage engine to replace RL-10	Aerojet & TRW (N.G. / ASS)	USAF RL									
CECE	Throttling LO_2/LH_2 for planetary lander	P&W, Rocketdyne	NASA									
J-2X	Upper stage for Ares then SLS	P&W, Rocketdyne	NASA	Still in work for SLS								
RS-2SE	Non-reusable SSME	P&W, Rocketdyne	USAF/NASA	Still in work for SLS								

spacecraft to dock with the ISS and return to earth for a straight forward safe ocean based recovery, also uses smaller pintle injector engines for all of its propulsive and control functions. However the Dragon capsule engine and/or thrusters use earth stable N_2O_4/mmh hypergolic propellants as compared to the LO_2/hydrocarbon engines used for all stages of the Falcon launch vehicles. Figure 9 shows a test firing of a Merlin Engine for Falcon and Figure 10 shows a Falcon 9 / Dragon capsule heading for the first commercial capsule docking at ISS. It is believed that similar engine design approaches will be used to power other new low cost boosters for several other commercial launchers that are currently in development.

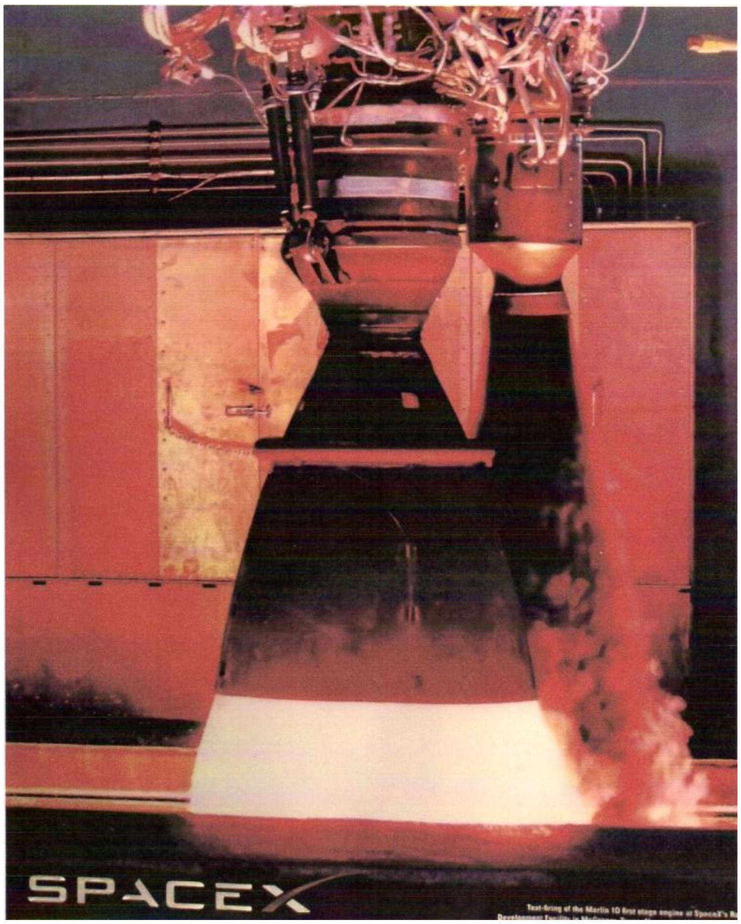

Figure 9: Test firing of a Falcon-9 Merlin 1st stage booster engine

However, it is very important to note that in order for U.S. commercial space companies to remain competitive on a world-wide basis, higher performance propulsion designs will probably have to be developed and incorporated into future commercial launch vehicles on a voluntary and low risk basis. Careful and successful integration of advanced propulsion technologies into U.S. commercial vehicles will allow these companies to sustain their leads as premier launch service providers all over the world. This is because introduction of these advanced higher performance rocket engines and other advanced technologies will enable the U.S. commercial companies to insert heavier payloads into space for lower weight and cost. This is also true for all types of commercial spacecraft where conventional on-board chemical propulsion technologies will largely be replaced by far more advanced systems such as all electric propulsion. Current space systems using on-board chemical propulsion that are ultimately injected into GSO require about 50% of the injected mass to be on-board chemical propulsion. However, if an all electric propulsion system is used (with an I_{sp} that is 10 to 15 times greater than the best chemical system) the ratio of injected spacecraft payload to the on-board propulsion system goes from 50-50 to 80% useful payload.

Figure 10: Lift-off of the first commercial Falcon-9 / Dragon capsule to dock at ISS

This is exactly where support from the U.S. government sponsored advanced propulsion technology development programs is needed to advance commercial programs with successful government industry partnership using "Space Act" type agreements or contracts, so that the U.S. commercial space industry remains highly competitive.

8.0 Conclusions

A good example of advanced propulsion technology that should be developed by the U.S. government for introduction into future U.S. commercial launch vehicles is an advanced closed cycle, oxidizer rich stage combustion (ORSC) engines, that are already in use in several other countries, especially in Russia (in some cases for decades). The newer, higher performance U.S. version of an ORSC engine would be greatly enhanced through the utilization of advanced materials and fabrication techniques and the designs themselves would be generated using the latest advanced CFD and other highly capable analytical model codes and tools that would be used to maximize performance, durability/life and help to eliminate any combustion instability issues while reducing the amount of traditional and expensive excessive testing needs relied upon in the past as discussed earlier in this paper. Other systems technology advancements, under government sponsorship, must also be achieved in order to take full advantage and completely integrate new higher performance propulsion system technologies and the associated rocket engines and/or thrusters into newer vehicles. Some other high risk technologies, besides higher performance hydrocarbon fueled engines, that are recommended for government supported development to enable integration of the higher performance launch vehicle propulsion systems in a more reliable and effective manner are summarized in Table 7 below along with a statement of the function and purpose/need for each one.

TABLE 7. Some Examples of High Risk Technology Development Recommendations for Future U.S. Government Investment and Support

Risk Area	Risk Item	Function
ORSC Hydrocarbon Fueled Booster Engine	• Combustion Instability Resolution • Higher Performance • Power Balance • Physics-Based Analytical Predictive Models • Injector designs • Materials/coatings for O_2-rich environment survivability • Turbomachinery • Long life Bearings • Start and Shutdown Transients • Requirements for Vehicle Integration	Increase Launch Vehicle Performance and international competitiveness by decreasing launcher size and weight for the same payload (decrease glow)
Integrated Vehicle Health Monitoring	• Reliable/Robust Sensors • Real-Time Critical Decision Making / Data to take corrective actions • Identify and Develop Non-Destructive Inspection (NDI) options and quantify reliability prior to flight • System integration into asymmetric vehicle configuration	Reduce risk by detecting propulsion system failures and automatically correcting problem in flight in real time or pre-launch on the ground
Adaptive Guidance & Controls	• Integration with IVHM • Real-time Control Algorithms • Fast Response Actuators • Software Verification and Validation	In conjunction with propulsion system response, automatically correct any vehicle controls or flight mechanical stability problems
Advanced Higher Performance On-Board/In-Space Propulsion Technologies	• Higher Performance Electric Propulsion • Higher Energy Chemical • Nuclear Thermal/Fusion • Highly Radical/Advanced New Propulsion Physics	Reduce spacecraft size, mass, and travel times