2^{nd} Generation Reusable Launch Vehicle
2G RLV

Risk Reduction Requirements Program
2G RLV NRA 8-27: TA-3 & TA-4
Contract #NAS8-00168

Final Report
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TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

1.0 EXECUTIVE SUMMARY:

This NRA 8-27 study determined the inherently safest engine cycles, identified and evaluated technologies that enhance these cycles, and proposed a risk reduction methodology and program plan to incorporate these improvements in support of 2nd Gen. RLV architecture and program goals.

The study started by identifying six cycles that showed the potential for high safety and reliability, and that met the needs of the various 2nd Gen. RLV architectures. These cycles, which are described in detail in section 3.1.1, are:

- Dual Burner-Fuel Rich-Staged Combustion (DBFRSC)
- Dual Burner-Full Flow-Staged Combustion (DBFFSC)
- Single Burner-Fuel Rich-Staged Combustion (SBFRSC)
- Single Burner-Fuel Rich-Gas Generator (SBFRGG)
- Split Expander (SPLTEX)
- Single Burner-Oxidizer Rich-Staged Combustion (SBORSC)

A safety analysis was performed on each of these cycles based on the use of currently available state-of-the-art technologies, and a consistent Technology Readiness Level (TRL) of 7. This safety and reliability analysis is discussed in more detail in section 3.2, and a summary chart of Loss of Vehicle (LOV) rates is shown in Figure 1.

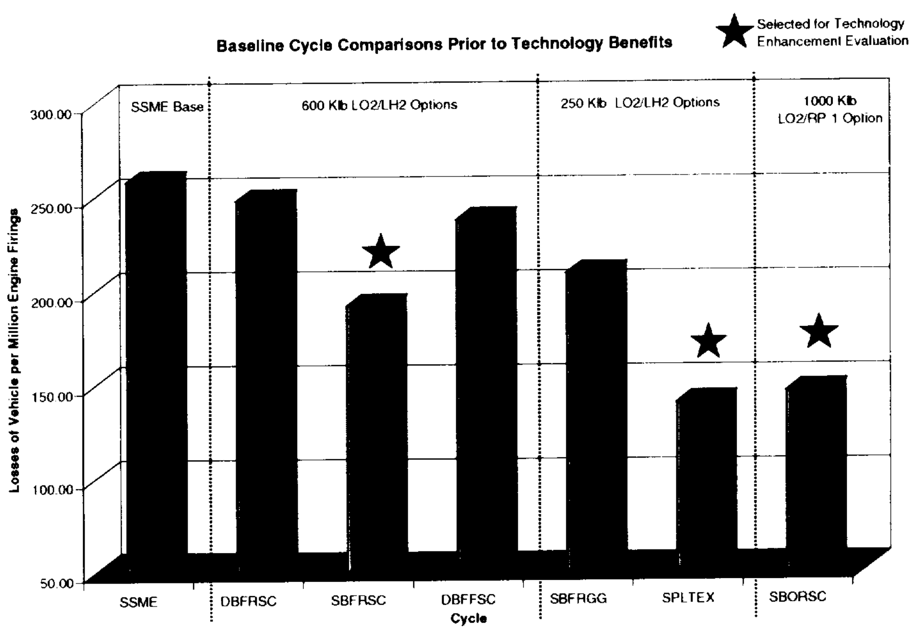


Figure 1 Engine Cycle Impact on LOV Rates

In addition to the safety analysis, cost and performance trade studies were performed and are documented in this report. Selection criteria were established, based upon program goals and objectives. Since flight safety is a primary objective of NASA, it is the primary selection criterion for this analysis.

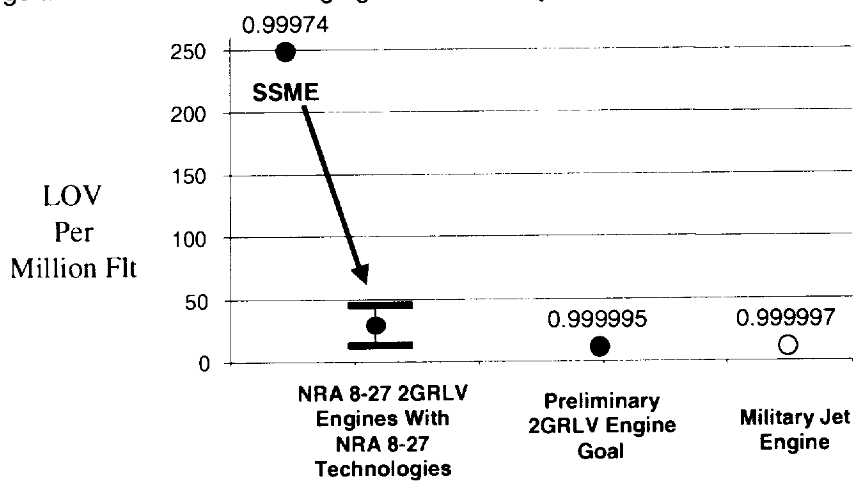
Three cycles, one from each thrust class, were identified as the inherently safest cycles, and selected for further study and analysis. These cycles are:

- Single Burner-Fuel Rich-Staged Combustion (SBFRSC)
- Split Expander (SPLTEX)
- Single Burner-Oxidizer Rich-Staged Combustion (SBORSC)

Next, technology improvements were identified and their effect on key parameters was studied (e.g. their benefit to LOV, production cost, etc). The various technology improvements, their applicability to the selected cycles, and their benefit to key parameters is discussed in sections 4 and 5. Key technologies include:

- Controller with Integrated EHMS
- Improved Durability Combustion Chamber
- Fail Safe Hot Gas System
- Material Containment System
- Gas Containment System
- Milled Channel Nozzle
- High Speed Main Fuel Pump

The selected cycles were evaluated with these technologies, and the resultant safety analysis showed significant improvement over the baseline (SSME). Based on the space Shuttle Quantitative Risk Assessment System (QRAS) data, the loss of vehicle (LOV) rate due to main engine is currently 258 per million missions. Depending upon the assumptions of effectiveness and implementation success of the technologies, the resultant safety of the engines ranges from 5 to 45 LOV events per million firings as shown in the following figure. This analysis is discussed in detail in section 5.2.



LOV Improvement Through Technology Improvements

A key piece of information that needs to be established is the "LOV per million" requirement for each booster engine. This requirement needs to support the NASA goal of system safety of 1 in 1000 LOV. It is our intention to work with the vehicle manufacturers to establish a reasonable requirement based upon their architecture, recognizing that the portion allocated to the propulsion system, as well as the number of booster engines, will change this value.

For the purpose of this study, we have established a goal of 5 events per million for a booster engine, which we believe is optimistic but achievable. As mentioned above, the level achievable by incorporating the current technology list is between 5 and 45 events per million, based upon the assumptions of effectiveness and implementation success of the various technologies. Contained within this report are the program plans to incorporate the various technologies as well as the overall system program plan. It is important to note that as the goal becomes more aggressive, the cost of the program increases, since additional technologies will need to be pursued to accommodate meeting a more aggressive goal. This report also contains a list of some "back-up" technologies that could be investigated.

In summary, this study has identified the three safest booster engine cycles, identified the technologies and program plans to incorporate these technology improvements, and shown that the resultant safety output should be in the correct range to support the architecture safety requirements.

2.0 INTRODUCTION

2.1 TA-3 & 4 OBJECTIVE

The objective of this NRA 8-27 study is to identify risk reduction areas that are applicable to several 2nd Gen. RLV architectures by performing cycle analysis and trade studies on applicable propulsion systems. Risk reduction activities were then identified to mature the technologies and cycles to production status.

This is the final report and addresses all of the work performed on this program. Specifically, it covers vehicle architecture background, definition of six baseline engine cycles, reliability baseline (space shuttle main engine QRAS), and component level reliability/performance/cost for the six baseline cycles, and selection of 3 cycles for further study. This report further addresses technology improvement selection and component level reliability/performance/cost for the three cycles selected for further study, as well as risk reduction plans, and recommendation for future studies.

2.2 APPROACH

Six propulsion cycles were chosen for this study. They are dual burner fuel rich staged combustion (DBFRSC), dual burner full flow stage combustion (DBFFSC), single burner fuel rich staged combustion (SBFRSC), single burner fuel rich gas generator (SBFRGG), split expander (SPLTEX), and single burner oxidizer rich staged combustion (SBORSC). These cycles were developed reflecting current technology levels (technology readiness level TRL=7). Studies were conducted on each of these cycles to characterize their performance and reliability with current state of the art technologies. The space shuttle main engine (SSME) reliability database (Quantitative Risk Assessment System – QRAS) was used as the baseline for these reliability studies, tailored to reflect the advantages and disadvantages of the cycles selected. The cycles were then ranked by LOV within each of the thrust classes required to meet the 2nd Gen. RLV architectures, see Figure 1. The cycle with lowest inherent LOV within each of the three thrust classes were then chosen for further study. A list of enhancing technologies that further mitigate the risks identified by QRAS was generated. These technologies were applied to each of the three cycles selected and the improvements to safety, reliability, performance, and cost were evaluated. The results of this study indicate that with reasonable and conservative analyses the improved engine LOV rate does not attain the desired goal of 5 LOV per million missions. The combined effect of the studied technologies achieves about 40 LOV per million. This result was reached using reasonable or conservative estimates of effectiveness. An optimistic reliability analysis assuming 95% effectiveness achieved a LOV of less than 5 per million. This indicates that the goal is achievable but that additional improvements should be identified to insure that a robust production propulsion system is delivered. A list of suggested areas for further study is located in the summary, section 6.

3.0 BASELINE ENGINE CYCLES

3.1 PERFORMANCE STUDIES

In an effort to help define a potential successor to the Space Shuttle, NASA recently commissioned a study of possible architectures for 2nd generation reusable launch vehicles (2GRLV). Known as the Space Transportation Architecture Study (STAS III), this project generated a wealth of new vehicle designs, and determined the direction for NASA's 2GRLV efforts. Vehicle companies were challenged to design systems that could significantly reduce the risk of loss of crew and vehicle, and cut the cost of putting payload in orbit. Safety and reliability were the major drivers in the STAS program, a fact reflected in this study and in ongoing work to define the 2GRLV.

A number of vehicle contractors responded to STAS with plans to meet the future need for reusable launch vehicles. Conceptual designs included shuttle derived/evolved, new single-stage-to-orbit (SSTO), two-stage-to-orbit (TSTO), and horizontal takeoff/horizontal landing (HTHL) vehicles. The wide variety of architectures proposed included an equally broad range of propulsion options, requiring different engine cycles and thrust sizes.

This study chose cycles which were representative of those incorporated in the STAS III reports, to encompass as many vehicle configurations as possible. Those liquid rocket engine cycles with the potential to satisfy the safety, reliability, performance, and cost requirements of the full spectrum of 2GRLV architectures were evaluated. A list of propulsion requirements was developed, focusing on thrust size and choice of propellant as the most important factors. The 2GRLV propulsion requirements have been defined and are shown in Table 1.

Engine propellants thrust sizes were chosen to fit the needs of the particular vehicles. For shuttle-derived vehicles, for example, the orbiter engines were estimated to use LH2/LOX propellants and produce approximately 600,000 lbf of thrust. Liquid boosters, which for shuttle-derived vehicles would perform the same function as the Solid Rocket Motors for the Shuttle, were specified as kerosene-fueled to satisfy the need for high thrust at lift-off. Thrust size was also estimated with consideration for abort modes. An important feature of the 2GRLV will be the ability to preserve the vehicle and crew in case of the loss of an engine. For this study, engine thrust size was chosen so that each vehicle would use multiple engines, reducing the impact of a single engine failure. While engine out capability is not currently a requirement, use of multiple engines allows safe return of the crew and the vehicle in the event of a single engine failure.

Technical parameters such as chamber pressure and inlet conditions were chosen to represent the current state-of-the-art among rocket engines. For the fuel-rich staged combustion cycles (FRSC), 3,000 psia, approximately the operating point of the Space Shuttle Main Engine (SSME) was chosen as the chamber pressure. The SSME inlet conditions were also used for evaluation of the FRSC cycles. Expander chamber pressures typically fall well below those of staged combustion engines, so for the split expander, a chamber pressure of 1,500 psia was selected as typical of a large expander using current technology. Preliminary system modeling was used to confirm this choice before the study began. For the single-burner fuel-rich gas generator cycle (SBFRGG), the STME (Space Transportation Main Engine) design was chosen as the best point of departure. Though the program was canceled before the engine was built, it better approximates the state-of-the art than existing gas generator engines that were originally designed decades ago. Lastly, the RD-180 engine was used as the starting point for the definition of the single-burner, oxygen-rich staged combustion cycle (SBORSC). The RD-180 operates at a chamber pressure just above 3,700 psia. The SBORSC chamber pressure was chosen at a somewhat lower 3,500 psia to accommodate the requirement for increased reliability in current rocket engine design.

Architecture Approach	Propellant	Thrust Size - pounds
Shuttle Derived/Evolved		
Orbiter	O_2/H_2	600K
Liquid Boosters	O_2/RP	800-900K
External Tank Propulsion	O_2/H_2	400-700K
New SSTO		
Orbiter	O_2/H_2	600k
New TSTO (Biamese)		
Booster Orbiter	O_2/H_2	400-700K
New TSTO (Other)		
Booster	O_2/RP	800-900K
Orbiter	O_2/H_2	200-400K
New HTHL		
Booster	O_2/H_2	200-300K
Orbiter	O_2/H_2	200-300K

Table 1 Architecture Approach and Thrust Class

3.1.1 Discussion of Baseline Engine Cycles

The following is a synopsis of the six baseline engine cycles chosen for study, including a brief description of how each cycle works and examples of current or planned engines using the cycles. More detailed performance information for each cycle, including chamber pressures, pump and turbine operating parameters and general system conditions, may be found in Appendix A.

3.1.1.1 Baseline Dual-Burner, Fuel-Rich Staged Combustion (DBFRSC) Cycle

The dual-burner, fuel-rich staged combustion cycle is a LOX/hydrogen engine with two fuel-rich preburners, one for each main turbopump, which create the hot combustion products used to drive the main turbines. The staged combustion cycle is typically referred to as a closed cycle, meaning that the turbine drive gases are retained in the system rather than dumped overboard, enabling high specific impulse when compared to gas generator engines. The staged combustion cycle is also capable of high chamber pressures, an additional performance benefit. In this cycle, the fuel is used to cool the main chamber and nozzle, while the LOX not introduced in the preburners is routed directly to the main injector. The Space Shuttle Main Engine (SSME) is an example of this cycle.

Various technical considerations influenced the design of this and other baseline cycles. Constraints imposed on this cycle included selection of fuel pump speed to remain within conventional bearing DN (diameter x speed) limits and design of the turbines to remain within AN^2 (Flow area x speed^2) ranges demonstrated on the SSME. In addition, fuel and LOX turbine temperatures, fuel pump tip speed and fuel pump exit pressure were constrained not to exceed SSME levels.

Detailed performance data for this cycle are summarized in Appendix A.1.

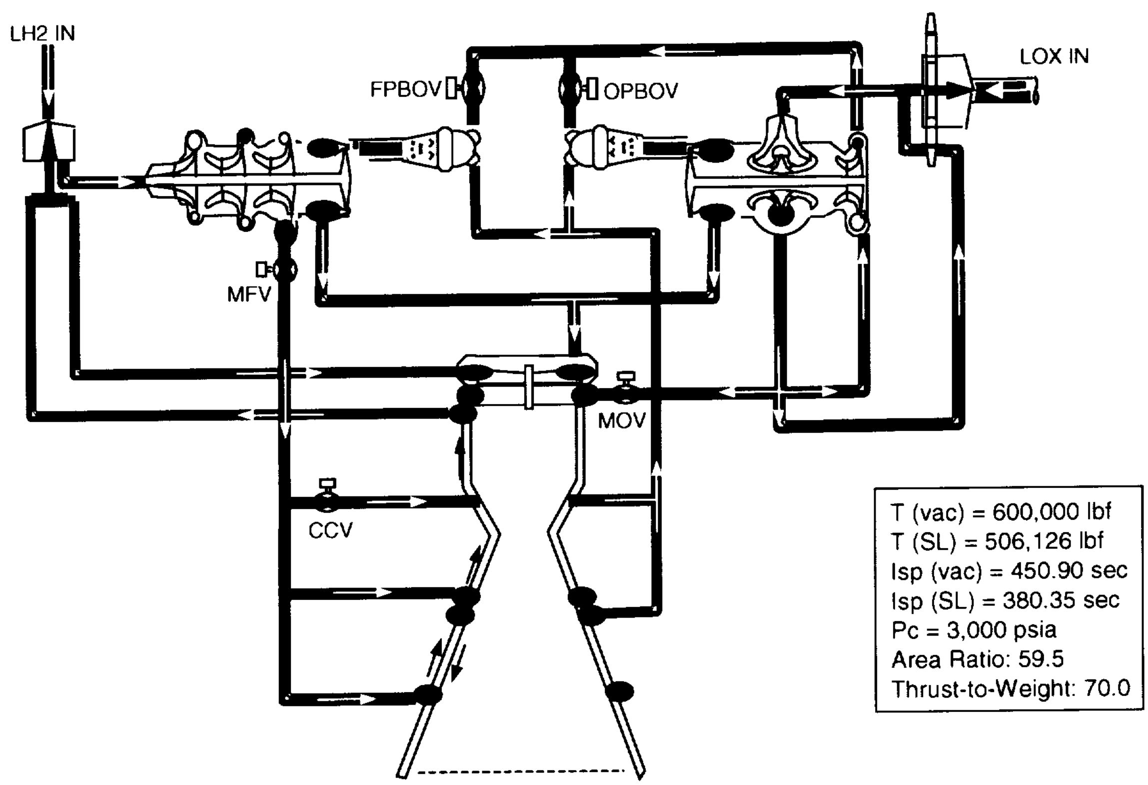


Figure 2 DBFRSC Cycle schematic

3.1.1.2 Baseline Dual-Burner, Full-Flow Staged Combustion (DBFFSC) Cycle

The dual-burner, full-flow staged combustion engine, also a LOX/hydrogen engine, differs from the fuel-rich staged combustion engine principally in that it uses all the flow entering the engine to drive the turbines, rather than just a fraction. The fuel turbine in this cycle is driven with a fuel-rich mixture of hydrogen and oxygen, while a LOX-rich hot gas mixture drives the LOX turbine. One principal advantage of the full-flow cycle is that it prevents unburned fuel and oxidizer from entering the same pump. Only hydrogen and products of combustion flow through the fuel turbopump, while only oxygen and products of combustion flow through the LOX turbopump. Another advantage is the lower turbine gas temperatures achievable with the full-flow cycle, enabling longer engine life and greater reliability. The projected RS-2100 design is an example of this cycle.

Constraints imposed on the baseline DBFFSC cycle included selection of fuel pump speed to remain within conventional bearing DN limits, design of the turbines to remain within AN^2 ranges demonstrated on the SSME, and prime reliable hot GOX compatibility in the Oxidizer preburner and turbine drive system. Fuel pump tip speeds and exit pressures also fell within SSME experience.

Detailed performance data for this cycle are summarized in Appendix A.2.

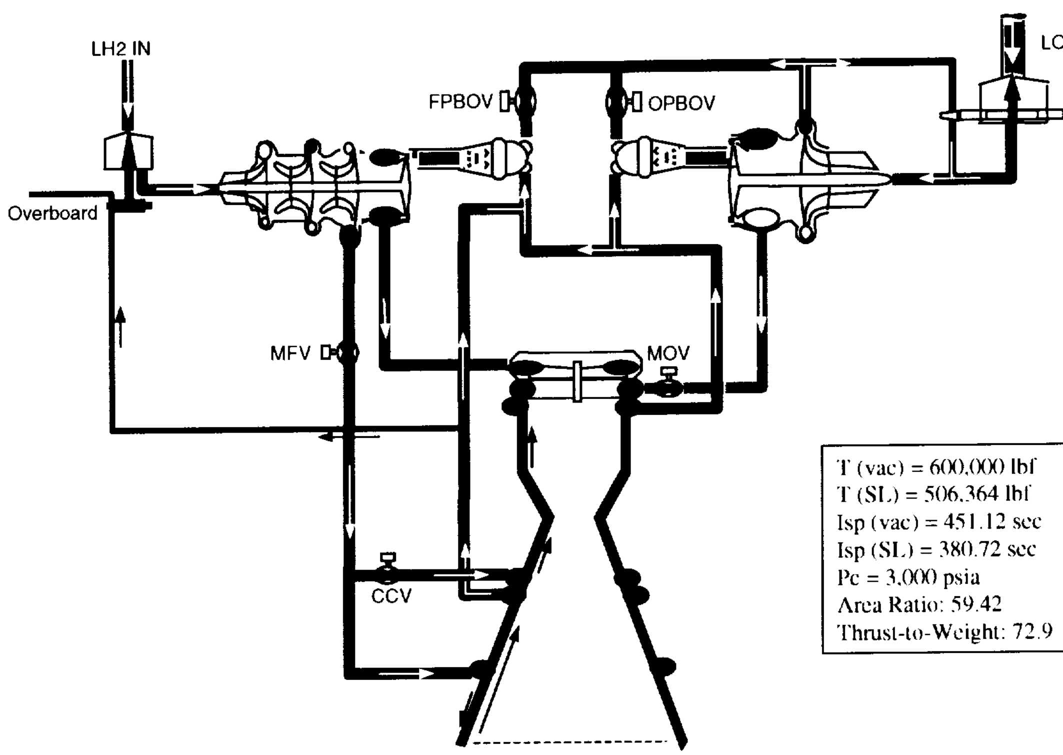


Figure 3 DBFFSC cycle schematic

3.1.1.3 Baseline Single-Burner, Fuel-Rich Staged Combustion (SBFRSC) Cycle

The single-burner, fuel-rich staged combustion engine eliminates the oxygen-rich combustion devices used in the dual-burner cycle, increasing engine reliability and safety. The hot combustion gases from the preburner drive both the hydrogen and LOX turbines before entering main chamber. While hydrogen fuel is still used to cool the main chamber and nozzle in this cycle, a portion of the fuel is directed to the preburner immediately after leaving the pump, reducing the pump load. This design decreases the turbine temperature, increasing engine life expectancy. In addition, the use of a single "liquid-liquid" preburner means that the high transient fuel turbine temperatures seen in the dual-burner staged combustion cycle do not occur. Additionally, the fuel and LOX turbine temperatures are essentially "averaged" in the single-burner system, allowing the peak temperature in the system to stay at a more benign level. The proposed COBRA engine is an example of this cycle. The Russian RD-0120 engine also uses this cycle, with a single shaft LOX and fuel turbopump.

Constraints imposed on the baseline SBFRSC included selection of fuel pump speed to remain within conventional bearing DN limits and design of the turbines to remain within AN^2 ranges demonstrated on the SSME. Fuel pump tip speeds and exit pressures were also maintained within state-of-the-art levels as defined by SSME.

Detailed performance data for this cycle are summarized in Appendix A.3.

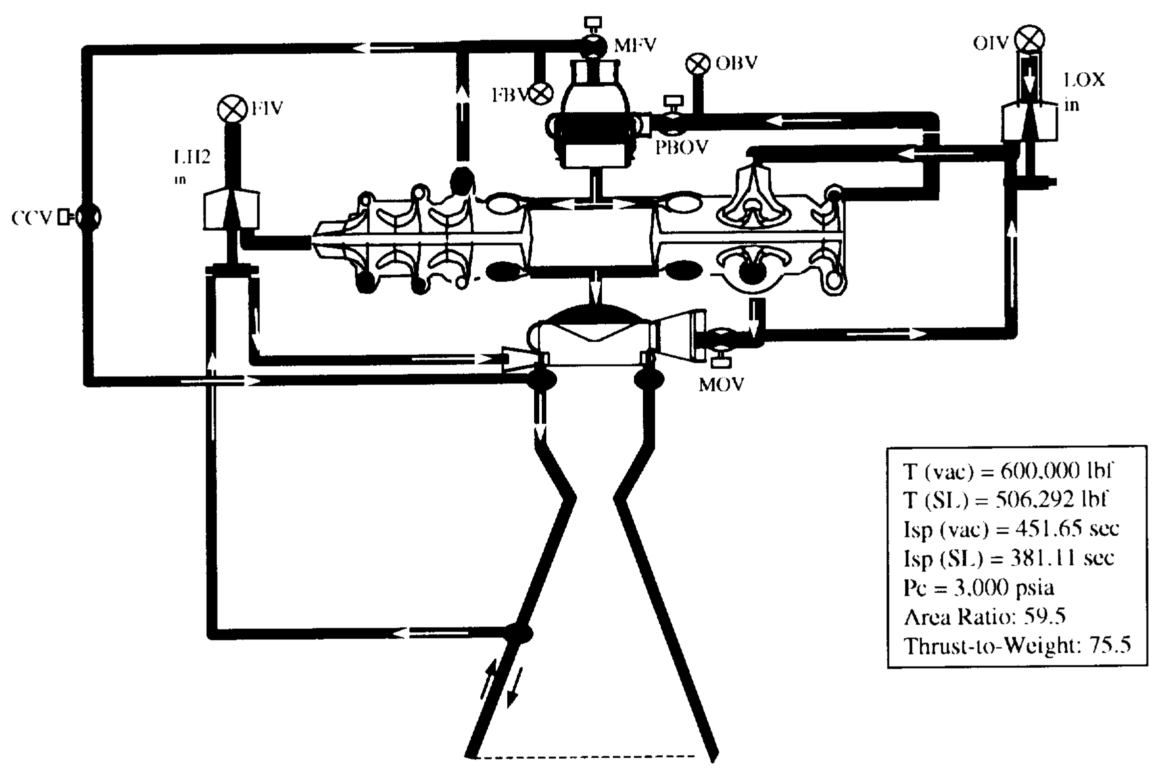


Figure 4 SBFRSC cycle schematic

3.1.1.4 *Baseline Single-Burner, Fuel-Rich Gas Generator (SBFRGG) Cycle*

The gas generator cycle utilizes hot gas created by the combustion of a small amount of fuel and oxidizer to drive the two main turbines. The turbine exhaust gas is dumped into the divergent section of the nozzle, rather than routed to the main injector, allowing for a much higher turbine pressure ratio and consequently less turbine flow. The advantages of the gas generator cycle include greater simplicity and generally lower cost compared to the staged combustion cycle, while the main disadvantage is the reduction in specific impulse caused by passing a smaller fraction of the total flow through the main combustion chamber. The Space Transportation Main Engine (STME), a booster engine design created in the early 1990's, is an example of this cycle. Existing gas generator engines includes the MA-5A used on the Atlas II and the RS-27A used on Delta II and Delta III.

Because system pressures and turbopump operating conditions are generally less challenging for gas generator cycles than for staged combustion engines, turbopump tip speeds, AN^2 and exit pressures for the baseline SBFRGG cycle did not exceed current technology limits defined by the SSME.

Detailed performance data for this cycle are summarized in Appendix A.4.

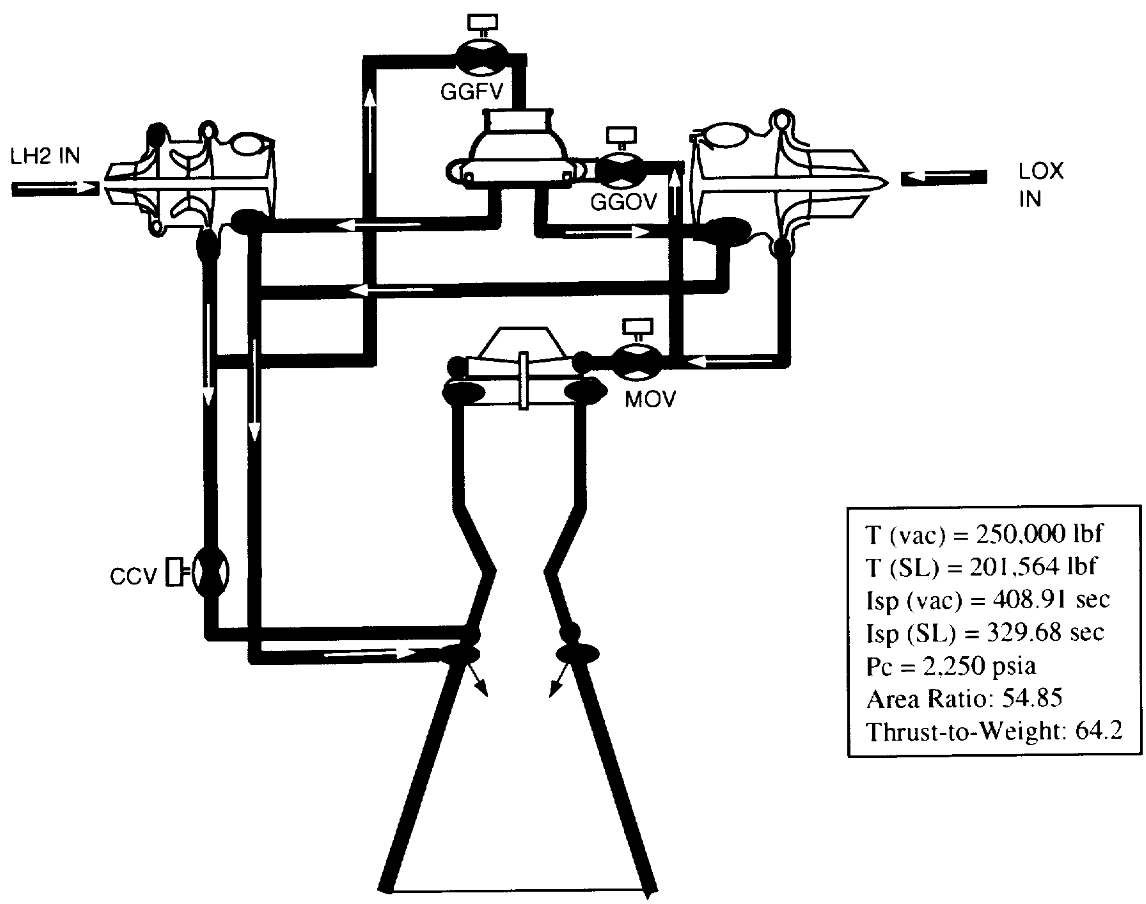


Figure 5 SBFRGG cycle schematic

3.1.1.5 Baseline Split Expander (SPLTEX) Cycle

The split expander cycle is driven by the heat absorbed by the hydrogen fuel as it cools the combustion chamber and regenerative nozzle. Rather than depend upon an additional combustion device to drive the turbines, the expander cycle uses the more benign process of heat absorption to gain the energy it needs to power the turbines. While this cycle allows a reduction in the number of combustion devices, heat transfer limitations restrict the chamber pressure achievable with an expander engine. In the split expander, fuel flow is split after the first stage of the pump, allowing some of the flow to be routed to the main injector while the rest continues on to the second stage of the pump. The advantage of the split flow is the overall decrease in pump horsepower, allowing a reduction in turbine work required. The proposed RLX engine is an example of this cycle.

Considerations for establishing this baseline included selection of fuel pump speed to remain within conventional bearing DN limits, use of a conventional single-circuit cooled chamber, and use of SSME-type chamber and nozzle heat transfer assumptions based on similar materials and configurations.

Detailed performance data for this cycle are summarized in Appendix A.5.

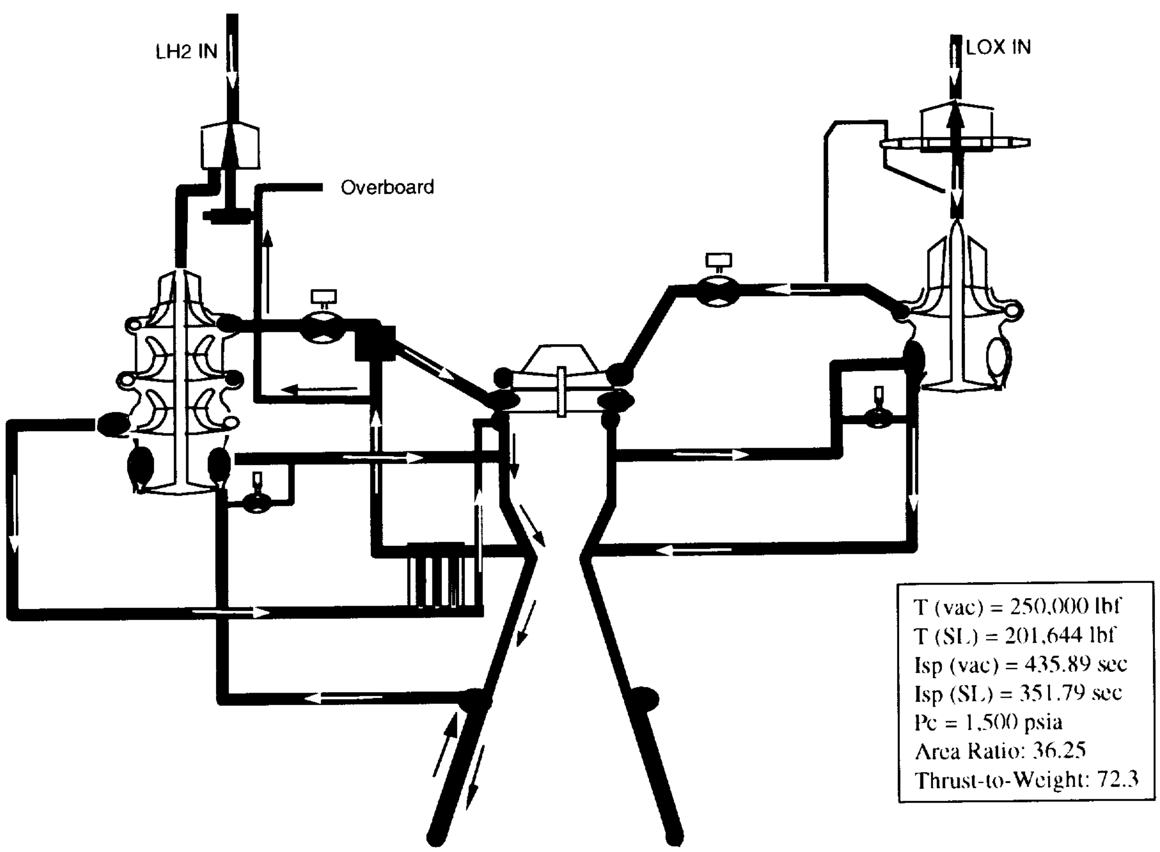
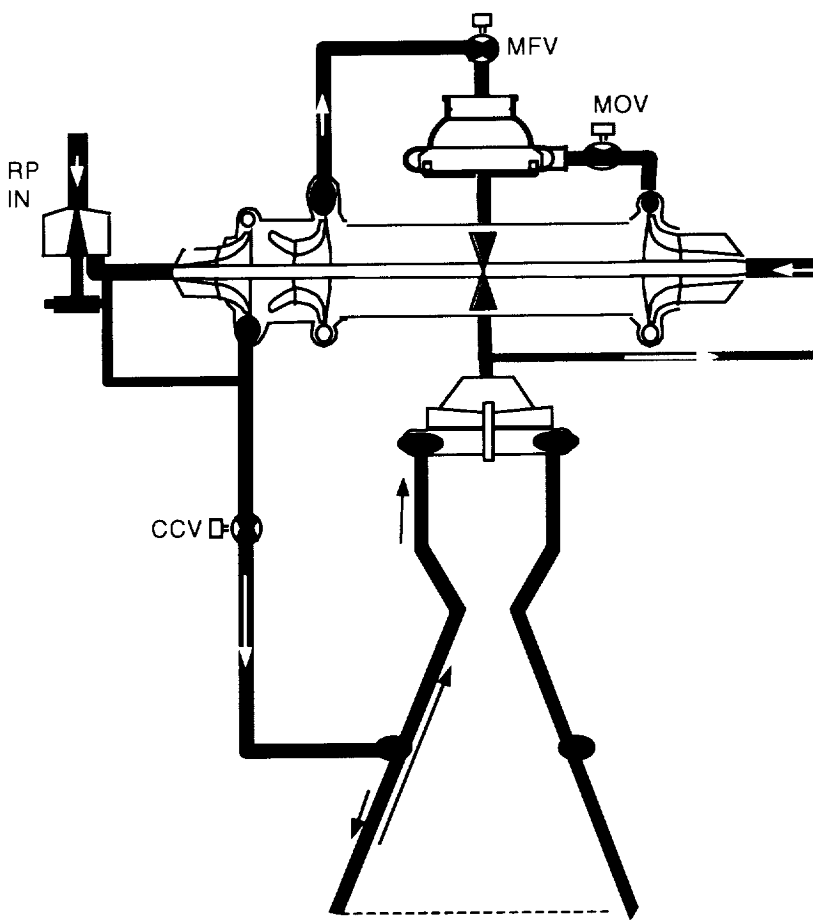


Figure 6 SPLTEX cycle schematic

3.1.1.6 Baseline Single-Burner, Oxidizer-Rich Staged Combustion (SBORSC) Cycle

The single-burner, oxidizer rich staged combustion engine burns a mixture of kerosene and oxygen. In this cycle, the preburner is run with a LOX-rich mixture of fuel and oxidizer. The reason for the LOX-rich operation is that more energy release can be obtained with a LOX-rich mixture of kerosene and oxygen than with a fuel-rich mixture burning at the same temperature. The LOX-rich preburner gases therefore contain more energy to drive the main turbine, allowing a lower turbine pressure ratio and ultimately a higher chamber pressure. Care must be taken, however, to ensure that rubs inside the turbine do not cause ignition of metal parts in the oxygen-rich environment. As with the fuel-rich staged combustion engines, the fuel is used to cool the main chamber and nozzle, and then is discharged through the main injector into the combustion chamber. While the kerosene fuel affords a lower specific impulse than hydrogen, it offers the advantage of greater density, smaller vehicle tank size, and storability at room temperature. The NK-33, a Russian engine derived from the engines used on the N-1 moon rocket, is an example of this cycle, as is the RD-180 used on the Atlas III.
The SBORSC baseline operated with a chamber pressure, turbopump tip speeds, DN's, AN^2's and system pressures well within ranges demonstrated by existing oxygen-rich staged combustion engines.

Detailed performance data for this cycle are summarized in Appendix A.6.



T (vac) = 1,000,000 lbf
T (SL) = 830,784 lbf
Isp (vac) = 354.06 sec
Isp (SL) = 294.15 sec
Pc = 3,500 psia
Area Ratio: 75.47
Thrust-to-Weight: 74.0

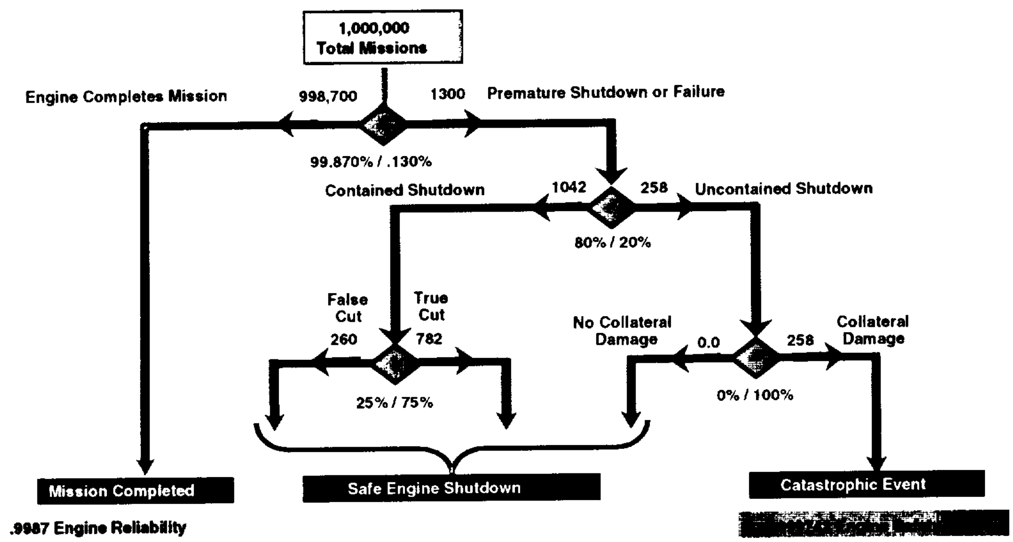
Figure 7 SBORSC Cycle schematic

3.2 RELIABILITY STUDIES

3.2.1 Background and Study Basis

NASA's primary goal is safe propulsion for robust RLV architectures. The NASA stated safety goals for the 2nd Gen RLV are LOV < 1 in 1,000 (~4X improvement over Shuttle) and Loss of Crew (LOC) < 1 in 10,000 (~40X improvement over Shuttle). Per the NASA QRAS, current (Baseline) LOV rate due to main engine is 258 per million as shown in Figure 8.

RLV Baseline Per SSME



2GRLV Safety Improvement Targets

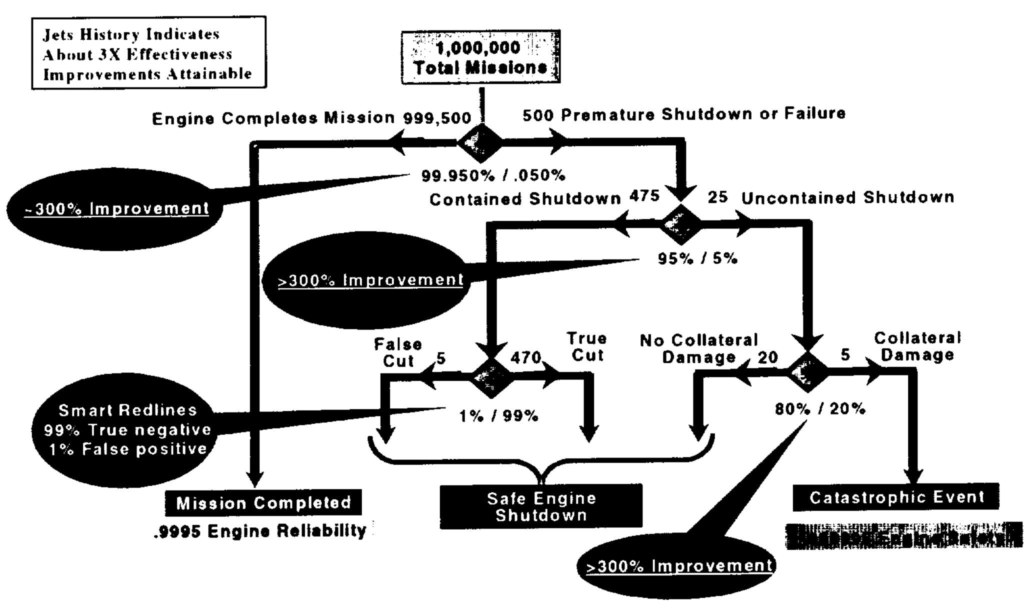


Figure 8 Safety: Baseline and Improvement Target

Figure 8 also shows a reliability tree approach to achieving a 5 per million LOV, based upon P&W's experience with improving jet engine reliability, which indicates that it is possible to achieve an improvement of at least 300% on each level in the tree. Based on QRAS, we currently estimate that 0.13% of missions will result in a shut down. With approximately a three-fold improvement we expect only 0.05% of missions result in a shutdown. Likewise, for shutdowns, currently approximately 20% of shutdowns are expected to result in an uncontained event. Through our jet engine experience we can expect to achieve a 5% uncontained event rate. Currently we assume 100% of uncontained events will cause collateral damage and result in loss of vehicle. New technologies and vehicle architecture improvements will allow us to contain 80% of these types of failures. This brings us to a goal of 5 failures in a million missions. This represents a catastrophic event goal of 5 per million missions (or 0.999995 catastrophic reliability). This study will determine the effectiveness of the cycles selected and application of new technologies to meeting this goal.

A three-phase approach was employed in this reliability analysis to go from the baseline SSME QRAS data to the projected improvements for the 3 cycles selected with new technologies incorporated. To keep track of parameters a primed system was used. The key below describes the failure rate (Rf) parameters and their designation depending on phase.

Phase I – Rf*, Rs*, Ruc*, Rm*;	SSME baseline rates
Phase II – R′f, R′s, R′uc, R′m;	Rates for six baseline cycles
Phase III – R″f, R″s, R″uc, R″m	Rates for 3 selected cycles, with Technology benefits incorporated

*Defined in Table 2

A study flow chart is shown in Figure 9. The study started by selecting a base cycle. The SSME was selected for the base cycle largely due to the comprehensive QRAS data available for the SSME. Next, this baseline data was adjusted to account for differences in the six engine cycles, and lessons learned to bring each cycle to a TRL=7. The differences between cycles included component environment, configuration and inherent cycle operation. After establishing an equivalent baseline, the cycles were divided into thrust class where the cycle with the lowest inherent baseline risk was selected for further review. Finally, a list of enhancing technologies selected to mitigate the risks identified by QRAS was generated, incorporated into the chosen cycles, and a reliability assessment established.

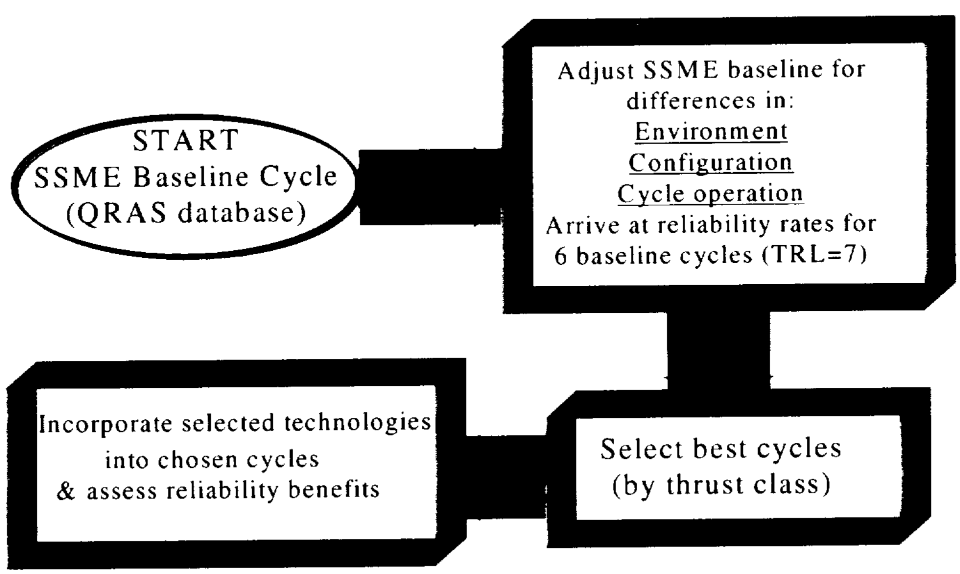


Figure 9 Reliability Process Flow

The NASA QRAS was used as the basis for failure rates in this study. Initial calculations revealed a hardware failure rate of approximately 1 in 4 missions, an engine shut rate of approximately 1300 per million firings and a LOV rate of 258 in a million firings as shown in Table 2.

	BASELINE SSME				
	(Rf) Failure Rate	(Rm) Maintenance Rate	(Rs) Shutdown Rate	(Ruc) Uncontained Rate	
** Mission Reliability (80% Confidence)	0.747460	0.747720	0.998590	0.999690	Engine Level Summary
Mission Reliability (50% Confidence)	0.748625	0.748883	0.998700	0.999742	
Total Rate Per Million Missions	251375	251117	1300	258	
COMPONENT					
Hot Gas Manifold	11283	11282	1.2	1.1	Component Level Rates
Comb. Chamber	95953	95902	52.0	51.2	
Main Injector	9496	9484	92.9	12.4	
Oxidizer Preburner	3593	3590	31.0	2.7	
Fuel Preburner	13909	13908	31.0	1.0	
High Press. Fuel Pump	47683	47607	185.9	76.4	
High Press.Oxid. Pump	27606	27561	62.0	44.7	
Low Press. Fuel Pump	17459	17457	1.5	1.3	
Low Press.Oxid. Pump	3073	3068	5.0	5.0	
Nozzle	13514	13483	154.9	30.8	
Heat Exchanger	6060	6058	2.0	2.3	
MCC Ignitor	295	295	0.0	0.0	
Fuel Inlet	40	40	13.8	0.1	
Oxidizer Inlet	40	40	13.8	0.1	
Fuel Flow Cntr.	40	40	13.8	0.1	
Oxidizer Flow Cntr.	40	40	13.8	0.1	
Fuel Pre-Brn	40	40	13.8	0.1	
Oxid. Pre-Brn	40	40	13.8	0.1	
Solenoid	40	40	13.8	0.1	
H2 Check Valve	40	40	13.8	0.1	
O2 Check Valve	40	40	13.8	0.1	
Fuel/Hot Gas System	37	24	31.0	13.1	
Oxidizer System	37	32	31.0	5.5	
Thrust Cntr.	0	0	0.0	0.0	
Pneumatic Control Sys.	111	103	92.9	8.5	
Controller(Electronics)	74	74	62.0	0.0	
Controller(Software)	74	74	62.0	0.0	
Control Sensors&Har.	148	148	123.9	0.0	
Hydraulic System	284	284	0.0	0.0	
Actuators	324	323	154.9	1.2	
** BASED ON 100,000 MISSION RUNS IN RELIABILITY MODEL					

Table 2 Baseline SSME Data

The present SSME reliability and failure modes are well documented in the QRAS baseline LOV. The components with the highest failure rate are the High Pressure Fuel Pump (HPFTP/AT), High Pressure Oxygen Pump (HPOTP/AT), main combustion chamber, and nozzle.

3.2.2 SSME Baseline – QRAS

Data for the baseline cycle was derived primarily from the NASA QRAS. QRAS is a joint effort between the NASA, Rocketdyne, Morton Thiokol, Pratt & Whitney and others, to model risks associated with the space shuttle vehicle. Both the component level hardware failure rates and the engine LOV rates were taken directly from QRAS. The engine level shutdown rates were also derived from the QRAS SSME LOV rates. Because it was beyond the scope of this study to perform a shutdown cause analysis, the NASA Safety & Mission Assurance (S&MA) office supplied the method for calculating an engine level shutdown rate from LOV rate. This method takes 20 percent of the shutdowns as resulting in loss of vehicle. The Rocketdyne reliability report to NASA was then used to distribute engine level shutdown rate down to the component level. The Rocketdyne report is submitted monthly to the NASA and was used as a source of current SSME reliability data for this study as well. Component maintenance rates were calculated from the hardware and LOV rates mentioned above. The component level LOV rates were calculated by distributing the engine level rate according to the component's percentage of the mean in the QRAS. QRAS lists "Other" risk in order to compensate for possible missed risks. "Other" was distributed among each component according to its percent weight. Table 2 shows the baseline results and identifies the rates as described below.

As stated above, most of the base hardware failure rates (Rf) were taken from QRAS initiating event data. We have addressed several components that QRAS had not addressed such as solenoid valves, hydrogen/oxidizer check valves, thrust control valve and controller. The valve failure rate values from QRAS were distributed evenly among all of the common valves for each cycle. The thrust control valve, applicable only to the SPLTEX, is shown with zero risk for the baseline, calculated from RL10 data, and inserted for that cycle only. Controller electronics and software risk rates were obtained from P&W military jet engine experience while control sensors/harness numbers were obtained from RL10 data.

The following summarizes the SSME baseline parameters and the basic formula used in their calculation:

<u>Component level rate formulas:</u>

Rf = Component level initiating event from QRAS
Rs = Engine Level rates distributed by percentages per the Rocketdyne reliability report
Ruc = Engine Level rate distributed per the mean percentage of each component
Rm = Rf – Ruc

<u>Engine level rate formulas:</u>

Rf = Σ (Component level rates)
Rs = Ruc*5
Ruc = Single engine LOV from QRAS FY 1999 Report
Rm = Σ (Component level rates)

3.2.3 Reliability For Baseline cycles

A primary requirement of this study was to compare a collection of different engine cycles against a common baseline. The baseline cycle was established from data obtained primarily from the NASA QRAS. The following six cycles (Table 3) were examined regarding differences in environment, configuration and inherent operation.

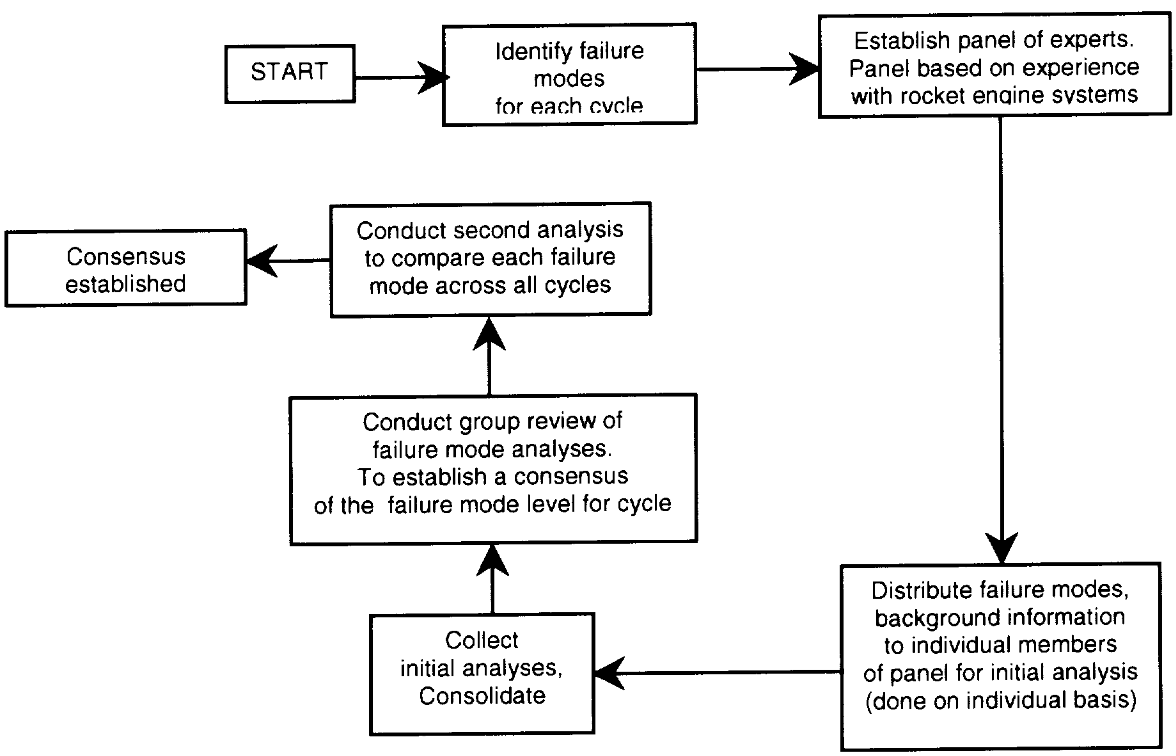
Description	Cycle
Dual Burner Fuel-Rich Staged Combustion	DBFRSC
Dual Burner [illegible]	[illegible]
Single-Burner Fuel-Rich Staged Combustion	SBFRSC
Single Burner [illegible]	[illegible]
Split Expander	SPLTEX
Single-Burner Oxidizer-Rich Staged Combustion	SBORSC

Table 3 Baseline Cycles Definition

Reliability estimates for all six cycles were anchored in the NASA QRAS for two reasons. First, a common basis must be used to ensure an unbiased method. Comparison of different engines using different baselines where some use a probabilistic assessment like the QRAS and others use demonstrated data like the RL10 introduces uncertainty in the results. Second, the effect of different engine thrust sizes can be accommodated through the use of a probabilistic assessment such as QRAS, whereas demonstrating data can not.

The baseline component level hardware failure rates (Rf) were analyzed and adjusted to accurately represent each of the six different engine cycles. Analysis of the six cycles was carried out in two parts. The first part reduced the baseline component rates to the failure mode level and used a Delphi teaming technique to estimate changes in a component's reliability caused by changes in environment and configuration. The Delphi team was made up of experts from Pratt & Whitney's Design Engineering, Structures & Dynamics, Propulsion Systems Analysis, Aerothermal, Controls Engineering, and Reliability & Mission Assurance (R&MA) organizations. Figure 10 shows the process flow of the Delphi technique. The second part used an event tree approach, based on LOV rates and failure event sequences, to further differentiate the six cycles (See Figure 12).

Figure 10 Delphi Type Process



A previous reliability study performed by Pratt & Whitney for the Rocket Engine Condition Monitoring System (RECMS) was utilized to divide each component of the baseline SSME into appropriate failure modes. Unsatisfactory Condition Report (UCR) data from that study was used to assign appropriate percentages to each failure mode. This division resulted in 316 failure modes being identified. The top 105 failure modes that contributed 90 percent of the overall engine risk were analyzed.

Parameters considered by the Delphi team were, temperature, pressure, speed, flow rate, configuration and material capability. Output from the technique was in the form of multiplier factors (Kf). These factors were then used to adjust the baseline SSME hardware reliability numbers obtained from the QRAS initiating event. A tracking database was established to track failure rate estimates obtained through the Delphi process. The sample database in Figure 11 shows the failure modes at left, the failure rate percentages (Kf), the starting SSME baseline value (in yellow), and the adjusted values, distributed by failure modes (R'f). Each cell that contains an "SSME BASELINE ADJUSTED BY Kf", also in yellow summed to a total engine level risk.

UCR ALLOCATES PERCENTAGE TO FAILURE MODES

DELPHI ESTIMATE FACTORS(Kf)

SSME BASELINE

SSME BASELINE ADJUSTED BY Kf

FAILURE MODES

DBFRSC	UCR %	UCR count	Kf	Failure Rate (R'_f)
MTBF(missions)				1.6
Engine Reliability(mission)				0.53292
System Risk				4.671E-01
Hot Gas Manifold				**2.955E-02**
P/B Exten. Liner Weld cracks	44.4%	8	0.96	1.261E-02
Weld or Material Failure	5.6%	1	0.97	1.593E-03
P/B retension sys. Failure	50.0%	9	0.70	1.034E-02
HGM Tot.		**18**		**2.455E-02**
Comb. Chamber				**2.513E-01**
MCC Liner Struct Failure	24.0%	37	0.98	5.918E-02
MCC Liner Surface Anomaly	35.7%	55	1.02	1.077E-02
MCC Liner Cool Channels Crack	8.4%	13	0.97	2.420E-03
MCC Housing Seal Failure	0.6%	1	0.98	1.881E-04
MCC Housing Structural Failure	1.9%	3	0.97	5.585E-04
MCC Housing Weld Anomaly	9.7%	15	1.00	2.879E-03
MCC Chamber Loss of Coolant	2.6%	4	0.98	7.523E-04
MCC Contamination	6.5%	10	1.00	1.919E-03
FRI erosion	10.4%	16	0.00	0.000E+00
Comb. Chamber Tot.		**154**		**7.866E-02**

Figure 11 Delphi Results Tracking Database

Once the baseline component failure rates (Rf) were adjusted through the Delphi to arrive at R'f, engine shutdown rates (R's) were calculated by maintaining the ratio R'f/R's = Rf/Rs. An assumption was made here that rocket propulsion system components, although operating under different conditions, perform comparatively similar functions. This being the case, it was assumed a component's shutdown rate (R's) was proportional to its hardware failure rate R'f. Uncontained engine failures (Ruc) at the component level were treated in a similar manner by keeping the ratio R'f/R'uc = Rf/Ruc. Maintenance rates were calculated by the equation Rm = Rf-Ruc. This suggests that the hardware rate (Rf) was all-inclusive and also that the portion of failures that destroys an engine would not result in a maintenance event. In light of the fact that maintenance rates were used primarily for costing purposes, an initial baseline calculation was performed as described and then a final calculation to assess technology impacts was performed in a similar manner. At this point Ruc for the six baseline cycles still reflected an SSME like Double Pre-burner Fuel Rich Staged Combustion Cycle. In Figure 12 a failure propagation tree approach was devised to highlight further differences between the six cycles.

To arrive at an accurate comparison between the six cycles, each was analyzed on a fundamental cycle operation basis. The analysis provided an additional level of fidelity in the comparison by identifying specific ways or modes in which each cycle could fail the major components involved and finally, failure consequences. In order to compare all cycles it was necessary to identify a complete set of failure paths that encompassed all cycle configurations under study. Once a complete set of paths was identified, visual aids were attached so those paths not applicable to a particular cycle could be eliminated. Figure 12 shows the initial tree, which was based on the SSME cycle and prior to any adjustments being made for the other six cycles. The trees labeled DBFRSC, SBFRSC, DBFFCS, SDFRGG, SPLTEX and SBORSC of appendix E are used to map failure paths and identify changes due to inherent cycle operation. The baseline component level LOV rates (R'uc) shown in Appendix E are input to the tree and a new engine level R'uc emerges after going through the analysis. Note that this new LOV rate is also denoted as R'uc.

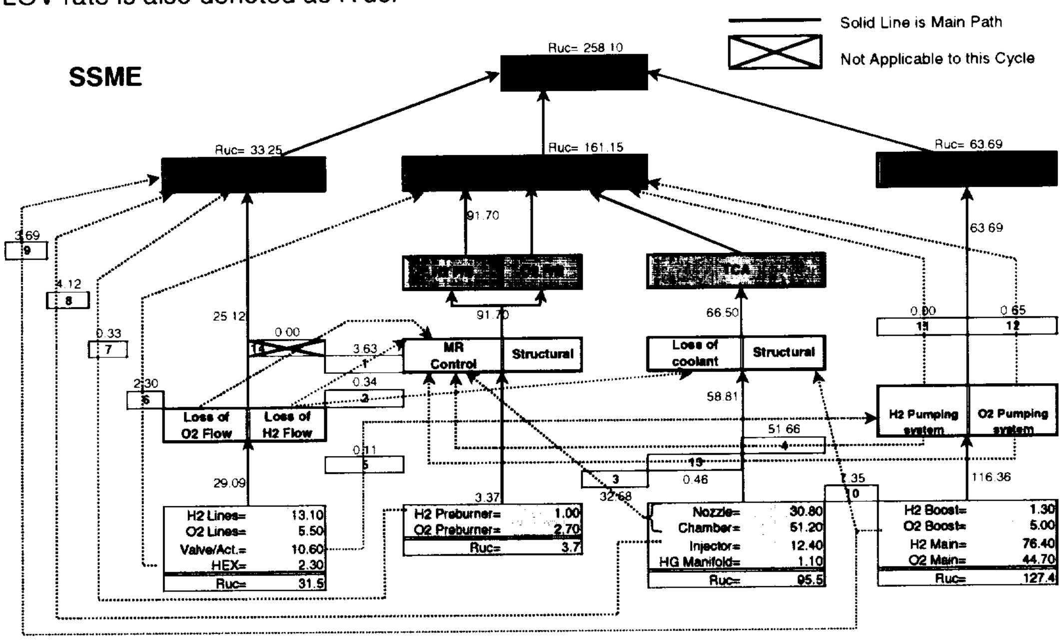


Figure 12 Initial SSME Failure Propagation Tree

Description of paths in Figure 12.

1. Hydrogen flow disruption can reduce hydrogen flow to the hydrogen-rich preburner(s) causing a damaging increase in mixture ratio.
2. Hydrogen flow disruption can starve main chamber coolant supply unless the chamber coolant flow is the predominate contributor to main chamber hydrogen flow.
3. For nozzle configurations cooled in series with preburner(s) a major nozzle leak can reduce hydrogen flow to the hydrogen-rich preburner(s) causing a damaging increase in mixture ratio. Chamber and nozzle leaks also result in loss of low pressure fuel turbo pump (LPFTP) function and loss of high pressure fuel turbo pump (HPFTP) turbine coolant.
4. If fuel and oxidizer turbopumps are not coupled, a severe mismatch can cause a damaging change in preburner mixture ratio.

5. If the fuel pre-burner oxidizer valve actuator (FPOVA) seizes during a pneumatic shutdown, the fuel pre-burner oxidizer valve (FPOV), main fuel valve (MFV) and chamber coolant valve will not close. This will result in a HPFTP over speed and subsequent Criticality 1 failure.
6. Damage to or failure of the heat exchanger (HEX) with mixing of gaseous oxygen (GOX) and fuel-rich hot gas results in burn-through of the hot gas manifold (HGM), high pressure oxidizer turbo pump (HPOTP) turbine or main injector.
7. Failure of propellant manifold welds causes release of non-combusted propellants into engine bay.
8. Oxidizer manifold/inlet or splitter failure in the main injector results in release of oxidizer into engine bay.
9. Pump housing failure results in release of non-combusted propellants into engine bay.
10. High-pressure turbopump turbine end damage may liberate material that impacts LOX posts and results in injector damage and structural failure.
11. Inter propellant seal (IPS) sleeve fracture results in loss of cooling and possible bearing and airfoil damage leading to case penetration.
12. High pressure pump housings and flanges can rupture releasing high pressure propellants into engine bay.
13. If fuel and oxidizer turbopumps are not coupled, severe mismatch can cause a damaging change in preburner mixture ratio.
14. Oxidizer flow disruption can reduce oxidizer flow to the oxygen-rich preburner(s) causing a damaging change in preburner mixture ratio.

To establish trees for the six cycles, a second Delphi comprised of R&MA and Project Engineering identified failure paths for each QRAS failure modes and P&W turbopump failure modes and effects analyses were used to establish the LOV percentages for each path identified. Since component uncontained failure rates were attached to each path, the method produced a high-level failure modes effects and criticality analysis. Figure 13 shows a simplified failure propagation tree with paths removed for explanation purposes.

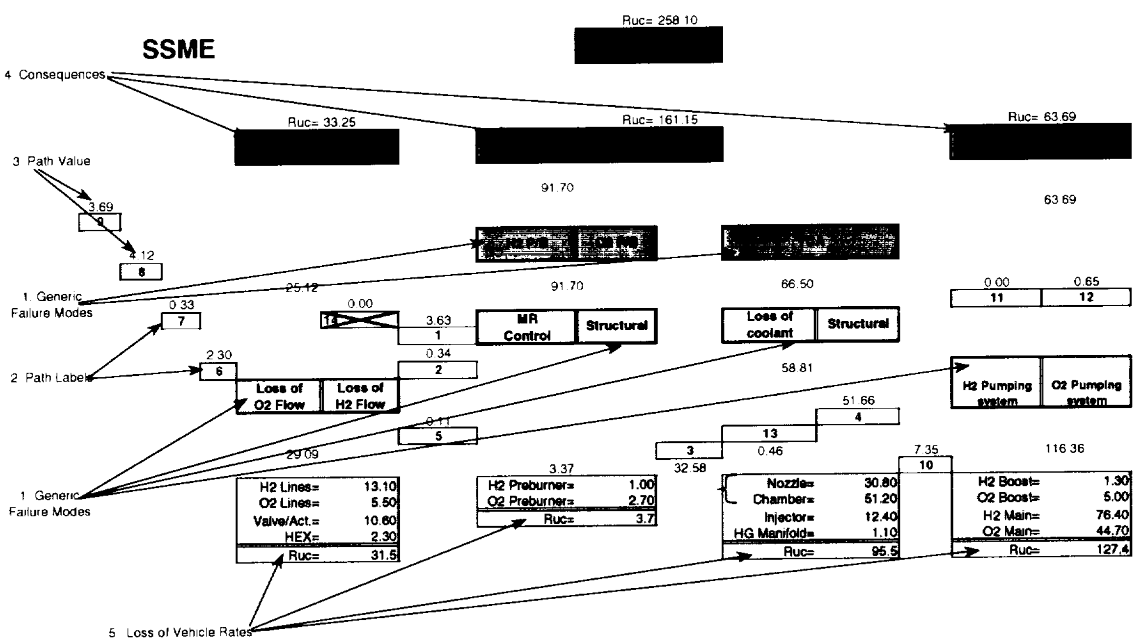


Figure 13 Schematic showing pieces of Failure Propagation Tree

Description of elements in Figure 13.

5 basic tree elements:
1) Generic Failure Modes - These failure modes were, in general, applicable to all cycles under review.
2) Path Labels – Path labels were used to identify and track descriptions of each path. An 'X' within the label box indicates the path is not a cause for catastrophic failures. Values that apply to the note are subtracted or mitigated by virtue of the cycle.
3) Path Values – Each path was assigned a value in failure per million firings based on estimated percent contribution. Two tools were used to estimate a path's percent involvement regarding the LOV rate in the lower boxes. The first was the QRAS event sequence diagrams. The diagrams describe failure events in sufficient detail as to provide a method which estimates the likelihood a component will fail in a manner that fits a particular path. Both turbopump Failure Modes and Effect Analysis (FMEA) were also used in a similar fashion.
4) Consequences – Failure consequences most likely to cause a LOV were identified.
5) LOV Rates – These rates were obtained for each of the six cycles based on the expected operational environment as described in section 3.2.3.

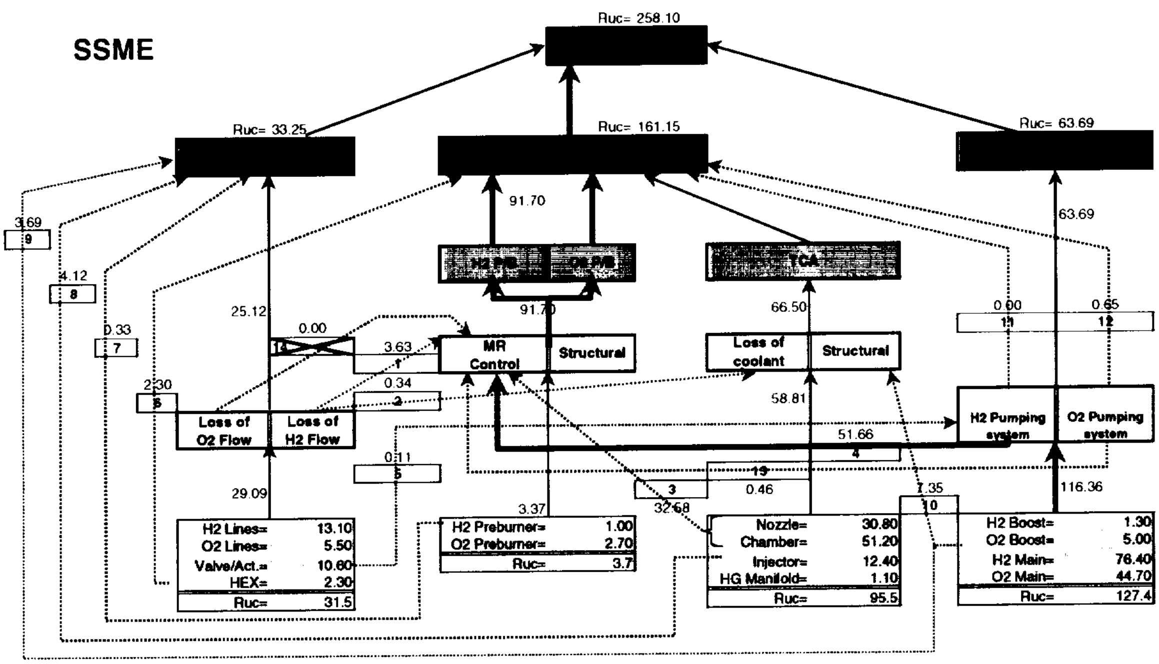


Figure 14 Example Tree with Path Highlighted

Figure 14 repeats the SSME baseline tree of Figure 12 with a path highlighted by bold lines for explanation purposes. As the tree indicates in the green box to the far right, the boost and high pressure pumps account for 127.4 LOV failures per million firings. Of the 127.4 failures, 7.35 failures are structural, 3.69 result in propellant leaks, and 116.36 proceed to pumping system problems. The path labeled 4 indicates that of the 116.36 failures, 51.66 are hydrogen pump related mixture ratio failures. The 51.66 combine with other mixture ratio and structurally related problems to result in 91.70 possible preburner failures per million firings. All 91.70 preburner related problems progress to combusted propellant leaks due to preburner burn-through events. The uppermost bold arrow indicates that 161.15 of the 258.10 LOV failures result from combusted propellant leaks. The red X shown through path 14 indicates that loss of oxidizer flow does not result in loss of vehicle, although it may result in an engine shutdown.

EXCEL was used to link the individual paths to another spreadsheet where failure modes were analyzed. Tracking of failure propagation was performed automatically within the tree and finally rolled up to an engine level rate.

The effect of transient and steady state temperatures on the turbopump turbines was considered at length during this study due to the harsh environments and high rotation speeds. The configuration of the pumps was assumed to be similar to the current HPFTP/AT and the HPOTP/AT. Speed, turbine temperature, and thermal transients (startup and shutdown) are known to cause the majority of life, and therefore, reliability concerns for the turbine end of the high pressure pumps. Specific problem areas were considered for improvement in order that gains in both reliability and life for turbine parts could be realized. The baseline engines designed with current levels of technology would not be exposed to the startup transient present in the SSME baseline. Compared to the SSME baseline the SBFRSC cycle turbine temperature is approximately 30% lower and pump speed is approximately 40% higher. Net effect was a 42% improvement in turbine hardware failure rate. Similar analyses of the other baseline cycles are as follows. The DBFRSC cycle with turbine temperature close to the SSME and speed increase similar to the SBFRSC resulted about a 2% improvement in failure rates. The SBFRGG cycle with only a moderate turbine temperature increase relative to the SSME base was debited approximately 5%. The DBFFSC cycle with a moderate decrease in temperature and a moderate increase in speed relative to the SSME received a benefit of approximately 30%. Although speeds and temperatures relative to the current SSME were lower, the harsh oxidizer rich environment prompted a debit of 20% for the SBORSC cycle. Due to its low turbine temperature the split expander received a benefit of 46% although fuel pump speed is high.

All of the baseline failure propagation trees are located in appendix E.

4.0 CYCLE SELECTION AND TECHNOLOGY IMPROVEMENTS

Figure 15 below shows the results of the reliability study of the six baseline cycles. The safest engine cycles are SBFRSC cycle in the 600Klb thrust class, the SPLTEX cycle in the 250Klb class, and the SBORSC cycle although alone in the 1000Klb class demonstrates good safety as well. These three cycles were chosen for further study.

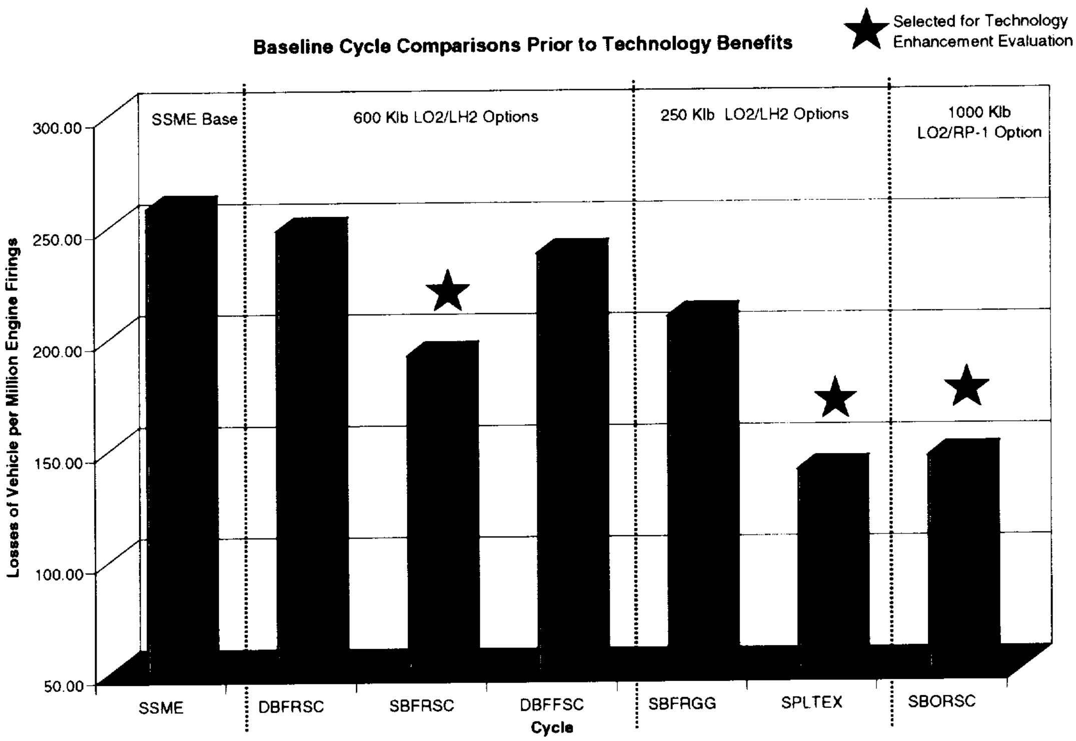


Figure 15 Comparison of Baseline Cycle LOV Rates

Safety, reliability, performance and cost trade studies were conducted on the three selected cycles to determine the effect of adding selected new technologies. Table 4 below shows the list of technologies that were chosen for these trade studies. The technologies were chosen because they show promise toward improving cycle safety/reliability/performance and can be developed in the next five years. The first eleven items are predominantly safety/reliability improvements and the last six are performance improvements. The table shows the applicability of each technology to each cycle. For instance the SPLTEX cycle does not have a preburner so preburner related technologies do not apply. Section 5.0 describes in detail the results of the trade studies.

Table 4 Configuration of each cycle with Technology Improvements

	TECHNOLOGY	SBFRSC	SPLTEX	SBORSC
1	Improved Durability Combustion Chamber	X	X	X
2	Fail Safe Hot Gas System	X		X
3	Milled Channel Nozzle	X	X	X
4	High Speed Main Fuel Pump	X	X	
5	Improved Durability Main Injector	X	X	X
6	Improved Durability Preburner Injector	X		X
7	LOX Cooled Nozzle Section *	X	X	
8	Electromechanical / Electro-pneumatic Actuators	X	X	X
9	Controller w/ Integrated EHMS	X	X	X
10	External Material Containment System	X	X	X
11	External Gas Containment System	X	X	X
12	Hydraulic Fuel Boost Turbine		X	
13	Low Oxidizer Inlet Pressure	X	X	X
14	Split Circuit Cooling		X	
15	Increased Main Fuel Pump Efficiency	X	X	X
16	Increased Main Oxidizer Pump Efficiency	X	X	X
17	Increased Combustion Efficiency	X	X	X

* This technology requires LOX boost pump turbine discharge mixing

5.0 CYCLE IMPROVEMENT STUDIES

5.1 PERFORMANCE STUDIES OF IMPROVED CYCLES

Once the new technologies to be studied were defined, those improvements that impacted engine performance were implemented in the engine cycle models. Each technology improvement was put into the engine model by itself, so that its impact on performance could be isolated without influences from other improvements. Important engine parameters such as specific impulse, pump exit pressure, turbine temperature and turbopump speed were recorded for each modeling run. The result of the study was a matrix showing how the baseline engine performance would change if any one of the proposed technology improvements was incorporated. Matrices for all three engine cycles included in this study are shown in Appendix B. Changes in pump speeds, turbine temperatures and system pressures are shown in the matrices, as well as design information such as pump DN and turbine AN^2. The parameters included in these matrices assisted in the creation of reliability estimates for the improved cycles, and helped guide the selection of the technology improvements included in the improved cycles.

Table 5 shows the impact of each technology on specific impulse for the three selected cycles. Most improvements, such as increased pump efficiency or hydrostatic bearings, did not change specific impulse because chamber pressure, engine mixture ratio and nozzle area ratio remained the same. Rather, these changes allowed decreases in pump exit pressure or turbine temperature, achieving the same performance as the baseline engines with more benign operating conditions. Some changes, such as an improved main injector, allowed specific impulse to increase without impacting engine operation. These changes essentially involve making more efficient use of propellants once they are delivered to the main chamber, and do not require changing pump speeds or turbine temperatures. Note also that thrust was held constant for all engines in this study, therefore, thrust does not appear as a variable in the following table.

	SBFRSC		SPLTEX		SBORSC	
Technology	Δ ISP, vac (sec)	Δ ISP, SL (sec)	Δ ISP, vac (sec)	Δ ISP, SL (sec)	Δ ISP, vac (sec)	Δ ISP, SL (sec)
LOX-cooled nozzle	0	0	0	0	N/A	N/A
Increased Main Fuel Pump Efficiency	N/A	N/A	N/A	N/A	0	0
Increased Main LOX Pump Efficiency	0	0	0	0	0	0
High Speed Main Fuel Pump	0	0	0	0	N/A	N/A
Low Oxidizer Inlet Pressure	0	0	0	0	0	0
Split Circuit Cooling	N/A	N/A	0	0	N/A	N/A
Increased Combustion Efficiency	+ 0.9	+ 0.9	+ 0.88	+ 0.88	+ 1.8	+ 1.8
Improved Durability FPL Chamber	0	0	0	0	0	0
Milled Channel Nozzle	0	0	0	0	0	0
Hydraulic Fuel Boost Turbine	N/A	N/A	+ 2.8	+ 2.8	N/A	N/A

Table 5 Performance Benefits for the improved cycles

5.1.1 Improved Single-Burner, Fuel-Rich Staged Combustion (SBFRSC) Cycle

The primary change to the operating mode of the SBFRSC cycle was the addition of a LOX-cooled nozzle section. LOX from the main pump exit is used to cool a small section of the nozzle just downstream of the throat. The resulting GOX is used to both pressurize the LOX tank and drive the LOX boost turbine. This arrangement permits the elimination of the heat exchanger now used on the SSME to pressurize the LOX tank, and consequently the elimination of a Category I failure mode. The use of GOX to drive the LOX boost turbine also decreases the amount of recirculation through the LOX pump. Ensuring good mixing at the boost pump exit is necessary to prevent gaseous oxygen ingestion into the main pump. These issues will be addressed in the design execution.

The improved SBFRSC cycle also includes a high-efficiency, high-speed fuel pump, incorporating hydrostatic bearings to achieve rotor speeds unattainable with conventional mechanical bearings. The higher speeds allow the elimination of one pump stage, reducing pump weight and improving the overall thrust-to-weight of the engine. These changes are also factored into the reliability study.

Other technology improvements impacting performance for this cycle include a milled channel nozzle, increased efficiency LOX pump, and lower LOX inlet pressure. Lowering LOX inlet pressure allows a reduction in tank pressure, lowering tank weight and increasing vehicle payload delivery capability. In addition, an improved main injector was assumed to increase combustion efficiency, resulting in a gain in specific impulse.

Detailed performance data for this cycle can be found in Appendix C.1.

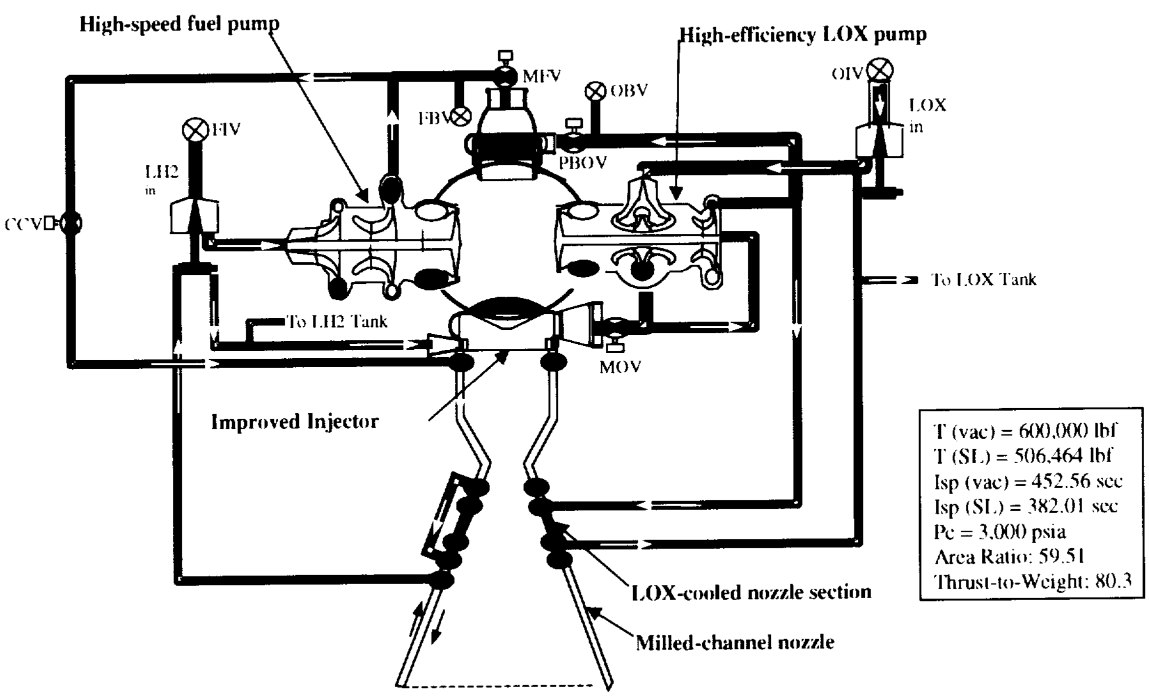


Figure 16 Improved SBFRSC cycle schematic

5.1.2 Improved Split Expander (SPLTEX) Cycle

Several performance-improving technologies were implemented for the split expander cycle. These changes included a high-speed fuel pump with hydrostatic bearings, operating at higher rotor speed than that achievable with conventional bearings. Higher rotor speed allows a smaller, more lightweight pump, that reduces overall engine weight.

The improved SPLTEX cycle, like the improved SBFRSC cycle, also includes a LOX-cooled nozzle section. GOX from this section drives the LOX boost turbine and pressurizes the LOX tank, eliminating the need for a heat exchanger to pressurize the LOX tank and reducing the amount of recirculation through the LOX pump. Another important change was the inclusion of split-circuit cooling. In this scenario, the chamber cooling circuit is divided into two sections, which are cooled in parallel rather than in series. The ensuing reduction in pressure drop across the chamber reduces horsepower demands on the main fuel pump, allowing a lower exit pressure which makes the engine more reliable and robust.

A hydraulic fuel boost turbine was also added to the SPLTEX cycle. The hydraulic turbine is driven by liquid flow split off from the first stage of the fuel pump. Turbine discharge flow is routed to the injector, where it is burned along with hydrogen from the main turbine. As a result, no hydrogen flow is lost overboard, unlike the baseline cycle, which uses a dump flow to power the fuel boost turbine. The increase in mass flow to the main chamber increases the engine's specific impulse by about 2.8 seconds.

Other technologies impacting performance for the SPLTEX cycle include milled channel nozzle, an improved injector (increasing combustion efficiency), a high-efficiency LOX pump, and lower LOX inlet pressure.

Detailed performance data for this cycle may be found in Appendix C.2.

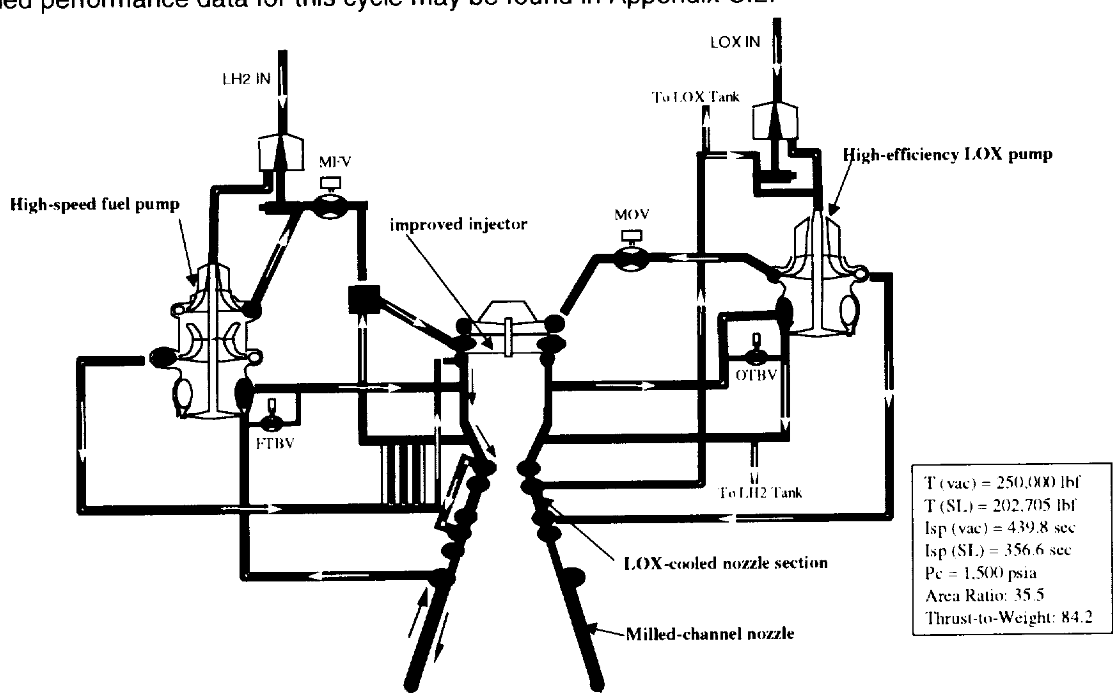
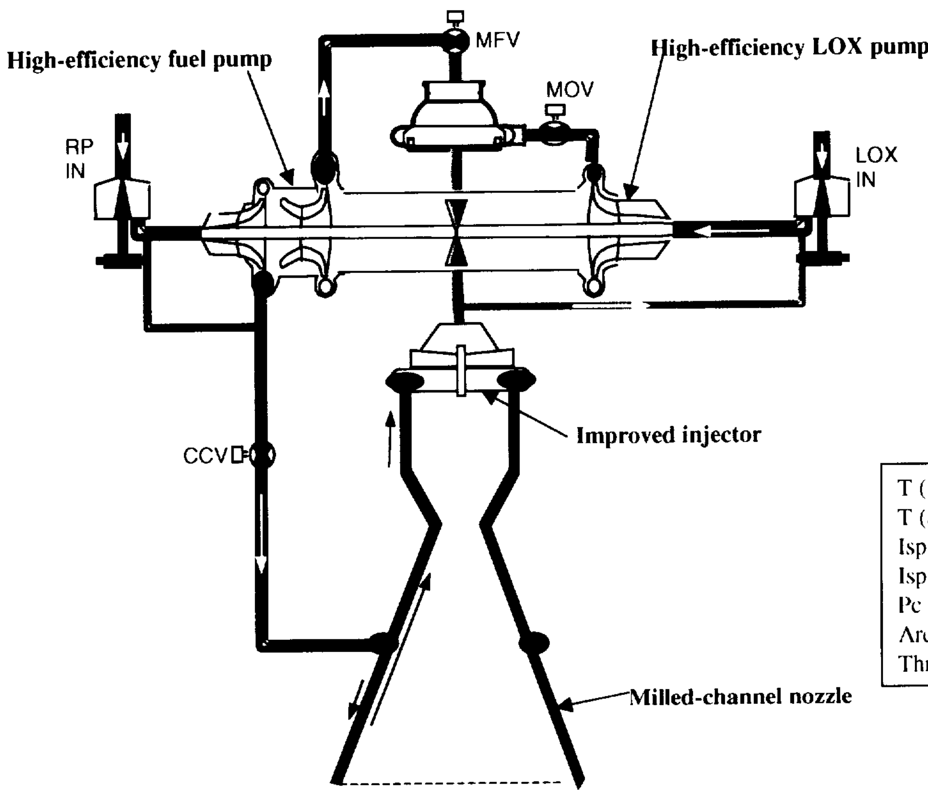


Figure 17 Improved SPLTEX cycle schematic

5.1.3 Improved Single-Burner, Oxidizer-Rich Staged Combustion (SBORSC) Cycle

The central performance-enhancing technologies improvements included on the SBORSC cycle were high-efficiency fuel and LOX pumps. High-speed fuel pumps such as those used on the improved SBFRSC and SPLTEX cycles are not projected for use on kerosene engines, but normal advances in turbine and pump design during the next five years will allow turbopump efficiency to rise relative to today's standards. The reduction in pump horsepower due to the more efficient pumps causes preburner pressure to drop, increasing robustness and reliability for the engine cycle. In addition to the pumps, the SBORSC cycle improvements included a milled channel nozzle, an improved injector to improve mixing and combustion efficiency, and low LOX inlet pressure.

Detailed performance information for this cycle may be found in Appendix C.3.



T (vac) = 1,000,000 lbf
T (SL) = 831,631 lbf
Isp (vac) = 355.86 sec
Isp (SL) = 295.95 sec
Pc = 3,500 psia
Area Ratio: 75.5
Thrust-to-Weight: 73.9

Figure 18 Improved SBORSC cycle schematic

Table 6 shows the changes in specific impulse, engine weight and thrust-to-weight for each of the three improved cycles with all technologies included compared to the baselines. Specific impulse increases for all three cycles, and the SBFRSC and SPLTEX show significant weight savings. The weight of the SBORSC increased slightly, due partly to the limited changes in turbomachinery that were made to that cycle. Both the SBFRSC and SPLTEX showed significant turbopump weight savings due to high-speed fuel pump technology included on those improved cycles.

Engine Cycle	Delta ISP (sec)	Delta Weight (lbm)	Delta T/W (vac)
SBFRSC	+ 0.91	- 474	+ 4.8
SPLTEX	+ 3.9	- 464	+ 11.9
SBORSC	+ 1.8	+ 24	-0.2

Table 6 Performance Deltas for Three Improved Cycles

5.2 RELIABILITY STUDIES OF IMPROVED CYCLES

A Delphi technique was used to estimate new technology impacts on the three selected cycles (SBFRSC, SBORSC and SPLTEX). Failure modes considered were those defined in the NASA QRAS. The QRAS failure modes were chosen because they represent the modes of greatest concern. QRAS failure modes considered can be found in Appendix D. The Delphi team for potential benefits reviewed each failure mode by considering mitigating effects of each technology. Estimation of technology benefits was performed at the failure mode level. The Delphi team estimated two factors, percent of problems addressed and percent effectiveness. The two factors were then multiplied to produce an overall benefit factor that was then applied to the QRAS risk.

The list below describes the technologies considered and expected benefits rated against the QRAS failure modes.

Milled channel nozzle - Tube leaks were eliminated since tubes were replaced by channels. Failure modes caused by steps, creases and bulges were eliminated due to the more robust design and the better heat transfer characteristics of the thinner hot wall possible with the mill channel process. Failure modes caused by braze voids were still applicable. Lessons learned were applied to aft end failures of the steerhorn, feed lines and manifold since feedlines will be placed at midsection, lowering transient loads, and design changes will be added to eliminate fillet and stubout failures. The additional plumbing required for the LOX cooled nozzle section resulted in an increase in fuel and oxidizer plumbing failure rates. Overall, the team estimated 80% of nozzle LOV failures will be addressed with 90% effectiveness.

Fail-safe hot gas system design - Of the three LOV modes for the hot gas manifold, two (cracks and failure of preburner retention system) were eliminated in the first Delphi. The remaining mode (Weld of parent material failure) risk was reduced 50% due to fewer welds and a more robust design. The new powerball's ability to successfully handle a preburner caused burn-through was estimated at 90%.

Improved Durability Main Combustion Chamber (MCC) with cast structural jacket - Of the 5 LOV modes, the two associated with flow recirculation inhibitors were eliminated, (FRI erosion and delamination of nickel plating). One was still applicable (Aft manifold weld failure). Cold wall leaks were less of a problem due to lower thermal fatigue resulting from better heat transfer. The last failure mode (Outlet elbow cast surface failure) was mitigated through changes to materials and processes. Fifty percent of the TCA problems (liner/Jacket welding) addressed with 90% effectiveness.

Improved Durability Preburner and Main injectors – Improved durability is achieved by incorporation of one-piece platelet fabrication. Of three LOV failure modes one remained (Rupture of Oxidizer Manifold). The second (Interpropellant plate anomaly) was estimated at 90% effective. The third (Heat shields impacted by FOD) was eliminated since LOX posts were eliminated. Overall it was estimated that 70% of problems were addressed with 95% effectiveness.

LOX Cooled Nozzle Section as heat exchanger - This feature eliminated a current criticality 1 failure but also adds concerns to the oxidizer system since new manifolding was needed. Failure of the new GOX system could cause LOX boost pump shutdown and engine shutdown. NASA testing has shown that moderate LOX leaks are not a problem.

High Pressure Fuel Turbo-Pumps - The high speed fuel turbopumps feature hydrostatic bearings, unshrouded pump impellers, and integral rotor assemblies (impellers, shaft, disk and turbine blades). Parts count reduction benefit is 30% for the SBFRSC and 60% for the SPLTEX. The overall reduction in failure rate was estimated at approximately 20%.

Controller with Integral Engine Health Management System - The in-flight fault detection and accommodation are integral to the controller. The diagnostics, prognostics and health monitoring functions are accomplished in the EHMS. This provides both a real-time Fault Detection and

Accommodation and prognostic benefits for vehicle safety. Self-Tuning On-board Real-time Modeling (STORM) was also assumed to be a part of the total Control/EHMS package. STORM can reduce false responses due to malfunctioning sensors by providing synthesized sensor readings, providing a basis to determine sensor validity prior to responding. Additionally STORM can provide simulated sensor input, based on other measured and calculated parameters, to maintain engine functionality in the event of sensor failures.

The EHMS system allows accommodation of system failures and is very effective in reducing false shutdowns through incorporation of redundant systems and real time on board modeling. EHMS can also effect safe shutdowns through effective detection and accommodation of potential catastrophic events. Effectiveness estimates of the EHMS system focused on items such as leakages of hot gas or uncombusted propellant and turbopump failures. For the turbopumps, problems involving the bearings, blades, housings and other rotating hardware are assumed detectable and accommodated by the EHMS through integral architecture with the controller. It was also assumed sensors needed to provide these input were available and adequate. The study used a conservative estimate of 50% for the above types of problems.

<u>Un-contained Gas/Combustion Products Containment -</u> These systems function to block or redirect hot gasses and uncombusted gasses. The systems will be constructed of fire proof or ablative materials. These systems may be combined with improved detection methods to further reduce LOV rates. Effectiveness of this new technology was assumed to be on the order of 75%.

<u>Material Fratricide Containment -</u> Addition of a high impact energy absorbing enclosure minimizes damage to other engines or vehicle systems by containing fragments after a catastrophic engine failure. Effectiveness was assumed to be 50% based on experience from the gas turbine engine industry. The gas turbine experience must be utilized in design of this system.

The above improvements were then applied to the propagation trees and a final LOV rate in failures per million was calculated. These failures were then mitigated through the combusted gas and material containment systems. Appendix F contains failure propagation trees for the three selected cycles.

Figures 19 through 21 give relative comparisons of LOV savings for each technology. These benefits transfer to the other parameters R"f (hardware failure rate) and R"s (shutdown rate) as well. The method of calculating R"f and R"s is similar to that described for R'f and R's. To calculate R"s the ratio R"f/R"s=Rf/Rs was held. This assumes that parts will still cause shutdowns but they will occur at a reduced rate as estimated by this study. Rm=Rf-Ruc was used here as well as in the baseline calculation for maintenance. This formula assumes Rf has captured all failure rates and hence all failures that do not cause LOV will result in a maintenance action. Tables in appendix F summarize results from the above calculations.

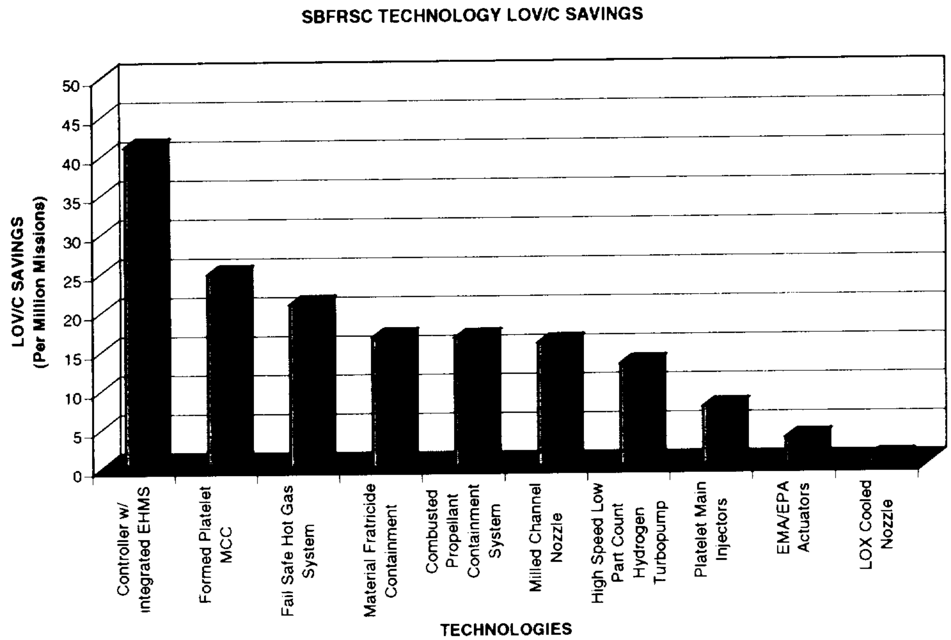


Figure 19 LOV rate for SBFRSC

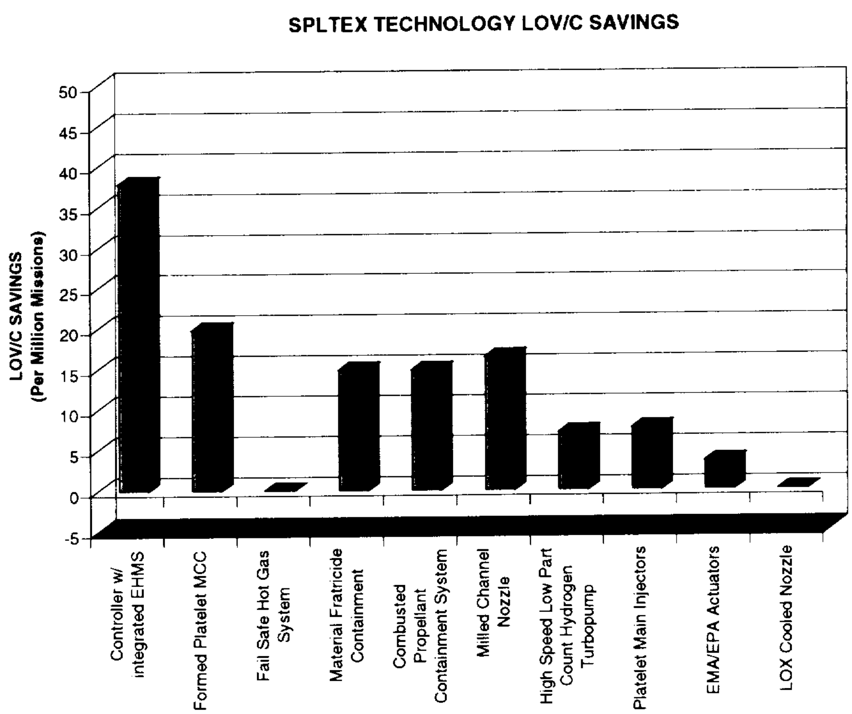


Figure 20 LOV rate for SPLTEX

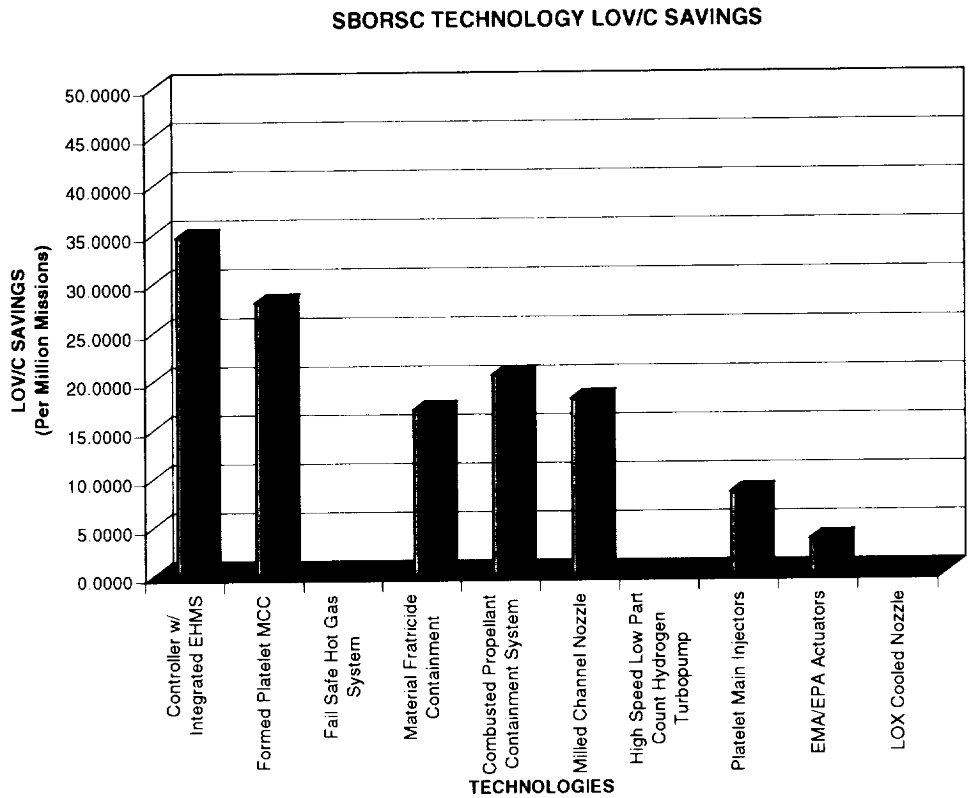


Figure 21 LOV Rate Benefit for SBORSC

	Each Technology Evaluated Separately Relative to Baseline	Δ SBFRSC	Δ SPLTEX	Δ SBORSC
	Baseline Engine LOV/C	Base	Base	Base
	Technology (Delta Reliability Relative to Baseline Engine)			
1	Milled Channel Nozzle	15.84	16.37	18.07
2	Fail Safe Hot Gas System	20.93	0.00	0.00
3	Improved Durability MCC - FPL	24.80	19.73	27.98
4	Improved Durability Injectors - FP	7.58	7.58	8.4
5	LOX Cooled Nozzle Section Replacing Gox Heat Exchanger	0.97	0.00	0.00
6	Electromechanical or Electro-Pneumatic Actuators	3.55	3.48	3.55
7	Avanced High Speed Main Hydrogen Pump (Unshrouded Impellers & Hydrostatic Bearings)	13.14	7.05	0.00
8	Controller with Advanced Engine Health Management System	41.13	37.83	34.67
9	External Gas Containment System	16.56	14.77	24.65
10	External Material Containment System	16.66	14.78	16.96
11	Increased Main LOX Pump Efficiency	0.00	0.00	0.00
12	Increased Main Kerosene Pump Efficiency	0.00	0.00	0.00
13	Hydraulic Fuel Boost Turbine	0.00	0.00	0.00
14	Low Oxidizer Inlet Pressure Capability	0.00	0.00	0.00
16	Split Circuit MCC Cooling	0.00	0.00	0.00
17	Increased Combustion Efficiency (Main Injector Improvement)	0.00	0.00	0.00
	All Technologies Together in Engine (Includes Interactions)	161.16	121.59	134.88

Table 6 Loss of Vehicle Benefit Summary

5.3 COST STUDIES

As part of the 2GRLV TA-3 engine studies, cost analyses were performed for each of the six baseline engine cycles. Development, production and operations cost estimates were generated for each of the engine configurations. After the baseline engine estimates were complete they were used as a basis for evaluating the cost impacts of the new engine technologies identified in TA-4 for possible use in the 2GRLV program.

5.3.1 Costing Approach

Cost estimates were generated for the six baseline engines using configuration information generated in the baseline engine cycle studies. Engine and component characteristics were defined for each engine and these definitions were used as a basis for the costs. All of the baseline engines incorporate existing technologies (TRL of 7 or above) and the component designs for these engines reflect these technology levels. Section 3.0 provides a description of the final configurations for each of the baseline engines.

In addition to configuration data, the operations cost analyses for the baseline engines used data generated in the reliability analyses for the baseline engines (See Section 5.2). Unscheduled maintenance rates were obtained from these studies and used for each baseline engine cycle.

To be able to assess the cost impacts of new technologies being considered for the 2GRLV program, cost estimates for the baseline engines needed to be made at a detailed level. A bottoms-up approach was used. Development, production and operations costs were generated at the component and/or activity level. After all elements were estimated individual costs in a particular cost category were summed to obtain total costs for the engine. All direct engine related costs (including propellants and government costs for testing at Stennis Space Center (SSC)) were included in the engine cost estimates; however, no NASA in-house costs for support of the 2GRLV Engine Program have been included. The cost estimates include all appropriate burdens but no profit or fee. All of the estimates are in constant FY2000 dollars.

Ground rules and assumptions used to estimate costs can have a significant effect on the magnitude of the costs defined. A set of costing ground rules and assumptions reflecting typical 2GRLV architectures and program plans was established for the study and used for the cost estimates. These costing ground rules and assumptions are discussed in the next section.

5.3.2 Cost Ground Rules and Assumptions

At the beginning of the TA-3 study a set of costing ground rules and assumptions was established for use in making the engine cost estimates. The costing ground rules included schedules and other programmatic information. They defined such things as quantities of vehicles and engines, mission usage rates and the engine development program and schedule. A 20 year period was selected for the operational flight program.

The intent of the ground rules was to reflect typical but notional 2GRLV architectures and programs. A separate set of ground rules and assumptions was established for each of the three different sizes of engines (250K, 600K and 1000K) being evaluated in the study. The final ground rules and assumptions used for the TA-3 and TA-4 cost estimates are shown in Table 7. These ground rules and assumptions were used for both the baseline engine cost estimates and the new technology cost assessments.

For the development cost estimates a notional engine development plan was needed to define the overall engine development effort. A 10 year engine development program that closely follows NASA's 2GRLV development plans was defined for use in the TA-3 and TA-4 engine cost studies.

This notional development plan is shown in Figure 22. During the first 5 years Concept Definition and Design and Risk Reduction activities occur. During the next 5 years Full Scale Development (FSD) and an overlapping one year Flight Test Program occur. The engine is only certified to a life of 20 missions in FSD. Certification of the engine to 100 missions is completed in a two year Life Extension Program that occurs immediately after FSD. This notional development plan was used for the baseline engine development cost estimates and for the technology development cost assessments.

5.3.3 Baseline Engine Production Cost Estimates

Engine production costs were the first cost items estimated for the baseline engines. This section discusses the methodology used and the production cost estimates obtained.

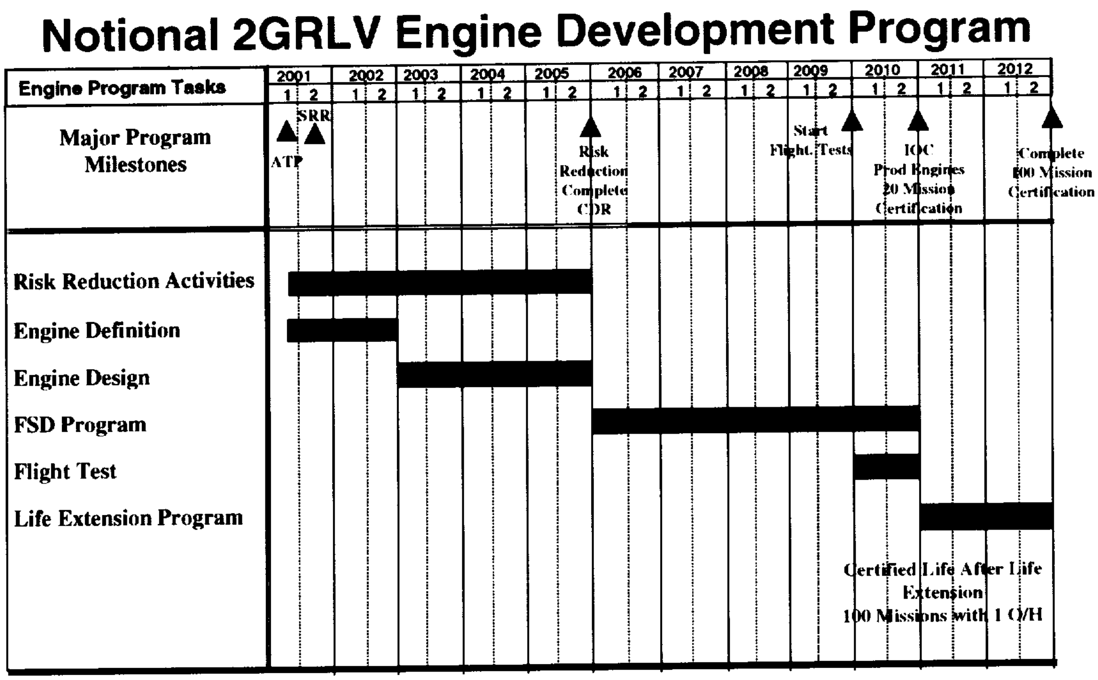
Production Cost Methodology

To make production cost estimates for the six baseline engines, component and engine characteristics were obtained for each of the engines from the cycle studies conducted for the baseline engines. This definition included the thrust size of the engines, the propellants used and flow routing of the propellants, chamber/nozzle cooling configuration, nozzle area ratio and size, number of pump and turbine stages and the quantity and location of control valves for the cycle. Differences in the type and quantity of components were also identified for the six cycles. Costs were then estimated for each component using this engine and component definition information. Table 8 shows the individual engine components for which production cost estimates were made

Table 7 Ground Rules and Assumptions

Engine Application	Manned Reusable Vehicle
Engine Usage - 250K	Booster Engines on Air Launched Vehicle
- 600K	Booster Engines On Shuttle Type Orbiter Vehicle
- 1000K	Booster Engines On Reusable First Stage
Propellants	Lox/LH2 for 250K and 600K; Lox/JP-8 for 1000K
Total Life Cycle Period	30 Years
Development Period	10 Years
Operational Period	20 Years
Number of Operational RLV's	5 Vehicles
Number of Engines per Vehicle	
250K Size	3
600K Size	3
1000K Size	2
Average Number of Missions per Year	20
Total Number of Missions Flown in Program	400 + 4 Flight Test
Number of Launch Sites	1 - KSC
Engine Development Program	
Risk Reduction Program Period	4 3/4 Years (Apr 2001 thru Dec 2005)
Engine Definition Period	1 3/4 Years (Apr 2001 thru Dec 2002)
Engine Design Period	3 Years (Jan 2003 thru Dec 2005)
Final Engine Design Complete Date	Dec 2005
FSD Program Period	5 Years (Jan 2006 thru Dec 2010)
Certified Life at End of FSD Program, Missions	20 Missions Without Overhaul
Flight Test Period	1 Year (Jan 2010 thru Dec 2010)
Number of Flight Test RLV's	1 (Becomes Operational RLV After Flight Test)
Number of Flight Test Engines (Including Spares)	
250K Size	4
600K Size	4
1000K Size	3
IOC Date	Jan 2011
Life Extension Program	
Life Extension Program Period (Additional Certification Testing)	2 Years (Jan 2011 thru Dec 2012)
Certified Life at End of Life Extension Program, Missions	100 Missions with 1 Overhaul
Time Between Overhaul (TBO)	50 Missions
Operational Program	
Operational Period	20 Years (Jan 2011 thru Dec 2031)
Number of Initial Engines Acquired (Excluding Flight Test)	
250K Size	18 (Including Spares)
600K Size	18 (Including Spares)
1000K Size	12 (Including Spares)
Engine/Component Attrition Rate (Other Than Life Retirement)	0.1 Equivalent Engines/Year
Average Replacement Engine Rate (After Initial Engines)	
250K Size	<1 per Year
600K Size	<1 per Year
1000K Size	<1 per year
Total Number of Engines Produced (Excluding Flight Test)	
250K Size	22
600K Size	22
1000K Size	16
Engine Delivery Rate	
250K Size	6/yr for 1st 3 yrs;<1/yr Thereafter
600K Size	6/yr for 1st 3 yrs; <1/yr Thereafter
1000K Size	4/yr for 1st 3 yrs; <1/yr Thereafter

Figure 22 Notional Development Plan Used for Engine Development



Historical cost information available from other engine programs was used to make the estimates for each component. Where necessary, adjustments were made for size, material and configuration differences. Initially production costs were estimated for expendable non man-rated engines since most of the historical data are for that type of engine. Once these costs were defined, adjustments were applied for the additional costs associated with producing reusable man-rated engines. These additional costs reflect additional redundancy and quality assurance activities required for reusable man-rated engines. Experience from the SSME turbopumps was used to help define the magnitude of this factor.

In addition to the engine component costs there are other cost elements that are part of engine production costs. These items include program management, engineering support, engine assembly, engine acceptance tests, propellants, deliverable data and packaging and shipping. Costs were estimated for each of these items and they were included as part of the engine production costs.

Historical cost data was used to estimate each of these items. It was assumed that the engine acceptance tests would occur at SSC and cost information obtained from SSC for the Space Transportation Main Engine (STME) and the Space Maneuvering Vehicle (SMV) engine programs was used as a basis for this estimate. Table 8 shows these other elements included in the engine production cost.

Initially engine production cost estimates were made for the 35th unit produced assuming a production rate of 6 engines per year. This unit was used as a reference unit because the production process would be established at this time and that unit is well below the steep portion of the learning curve.

Table 8 Typical Elements Included for Engine Production Cost Estimates

Turbopumps
Fuel Boost Pump
Oxidizer Boost Pump
Fuel Turbopump
Oxidizer Turbopump

Thrust Chamber Assembly
Main Injector
Main Combustion Chamber
Nozzle
Ignition System

Engine Controls
Engine Control Valves (4) - MFV, MOV, PBOV, CCV
Propellant Inlet Valves
Other Miscellaneous Valves (Cooldown, Purge, Drain, etc)
Control Valve Actuators (4)
Controller (Hardware and Software)
Hydraulic System (For Baseline Only)
Pneumatic System (Solenoid Valves and Plumbing)
Sensors
Cables and Interconnects
Engine Health Management System

Engine Propellant Ducting
Fuel Ducting
Oxidizer Ducting

Gas Generator and Hot Gas Systems
Preburner (1)
Ignition System
Main Case
Gox Heat Exchanger

Support Devices
Gimbal System
TVC Actuators (2)
Miscellaneous System Engine Hardware

Engine Assembly and Checkout
Engine Assembly
Packaging and Shipping to SSC
Post Test Checkout (At SSC)
Packaging and Shipping to Customer

Engine Acceptance Testing
NASA SSC Testing Costs
P&W/AJ Test Support
Propellants

Program Management and Engineering Support
Program Management
Engineering Support
Data and Documentation

Note: Elements Shown are for SBFRSC Engine Cycle

Learning curves were then used to calculate costs for the engine quantities shown in the costing ground rules and assumptions. Learning curve slopes of 90 to 95 per cent were used depending on the type of component or activity. No learning was applied to propellant and packing and shipping costs. The number of components manufactured in the development program for development and flight test engines was taken into account when establishing the unit number of the first production engine for the learning curve calculations.

Baseline Engine Production Cost Results

Production cost estimates generated for the baseline engines were one of the criteria used to down select technologies for further evaluation in the TA-3 and TA-4 studies. Additionally, the baseline production cost estimates provided a base for assessing the cost impacts of new technologies on the three engine cycles selected.

Because of the sensitivity of engine costs to many vehicle and capture assumptions, a generic and representative set of assumptions was made to run the models. The resultant production costs were normalized against the baseline, and the results reported as a delta from the baseline. While not providing an absolute production cost value, it does allow a good comparison of the projected production costs for the various cycles. The comparison provides more benefit than an absolute value in our current effort, because we are most interested in which are the most/least cost effective, whereas the absolute values can be misleading because they vary widely based upon the assumptions, and are not accurate unless we tie the absolute number to a specific vehicle and capture model.

In the technology assessment portion of this report, the impacts of the new technologies on engine production costs are shown as cost deltas relative to the baseline engine production costs.

The six cycles that were initially evaluated in the baseline engine studies consisted of three lox/hydrogen 600K thrust stage combustion engines, two lox/hydrogen 250K thrust engines (gas generator and split expander cycles) and one lox/hydrocarbon 1000K oxidizer rich staged combustion engine. Because of the different sizes only engines in the same thrust class are directly comparable. Relative comparisons of production costs for the 600K and 250K engines are shown in Table 9.

Engine Cycle Configuration		Relative Production Cost
600K Thrust Engines		
DBFRSC	Staged Combustion Cycle with Dual Fuel Rich Preburners	Base
DBFFSC	Staged Combustion Cycle with Dual Full Flow Preburners	102.0%
SBFRSC	Staged Combustion Cycle with Single Fuel Rich Preburner	96.40%
250K Thrust Engines		
SBFRGG	Gas Generator Cycle with Single Fuel Rich Gas Generator	Base
SPLTEX	Split Expander Cycle	94.0%

Note: 250K SBFRGG Cycle Does Not Have Boost Pumps

Table 9 Comparison of Production Costs for Baseline Engine Cycles

The three 600K baseline engines shown in Table 9 use different versions of staged combustion cycles. The DBFRSC engine is a fuel rich staged combustion engine using separate preburners to drive the fuel and oxidizer turbopumps. It is similar to the current SSME cycle. The DBFFSC engine is a full flow staged combustion engine using separate fuel rich and oxidizer rich preburners to drive the fuel and oxidizer turbopumps respectively. The SBFRSC is a fuel rich staged combustion engine using a single preburner to drive both turbopumps. For the SBFRSC the preburner and two turbopumps plug into a spherical main case which internally routes hot gases from the preburner through the turbines and into the main injector. Production costs for the three engines are shown relative to the DBFRSC cycle (current SSME). The dual preburner full flow staged combustion cycle (DBFFSC) has the highest production cost while the single preburner fuel rich staged combustion cycle (SBFRSC) has the lowest production cost of the three. The two full flow preburners (with additional control valve) cause the DBFFSC cycle to have the highest production cost. The single preburner (with one less control valve) and the compact main case account for the reduced cost of the SBFRSC engine. The lower production cost of the SBFRSC engine was one of the reasons it was down selected for further studies in this program.

The two 250K engines shown in Table 9 two totally different cycles. The SBFRGG is a fuel rich gas generator cycle engine while the SPLTEX is a split expander engine. The split expander engine has a lower production cost than the gas generator cycle engine. The SPLTEX engine includes boost pumps while the SBFRGG engine does not, making the production cost reduction for the SPLTEX more significant than indicated by the cost estimates. The expander cycle engine has a lower production cost because it operates with a cold turbine and does not have a separate burner to produce hot gases to drive its turbines. This lower production cost for the SPLTEX engine coupled with other cycle benefits was the reason this cycle was down selected for further studies in this program.

5.3.4 Baseline Engine Development Cost Estimates.

Engine development costs were generated for each of the baseline engines using the costing ground rules and notional development plan discussed in Section 5.2. This section discusses the methodology used and the development cost estimates obtained.

Development Cost Methodology

Development cost estimates were generated for the baseline engines using the schedules and development phases contained in the notional engine development plan shown in Figure 22. Since the baseline engines incorporate existing technologies, no risk reduction activities are needed for these engines. Development costs were estimated for the baseline engines for the Engine Definition and Design, FSD, Flight Test and Life Extension phases of the program. Each development phase was estimated separately and the resulting costs added to obtain total development cost estimates for the baseline engines.

To make the engine development estimates, a number of assumptions had to be made for the development program. These assumptions included such things as the amount of Design Verification testing, the amount of component testing, the quantity of new development engines, the quantity of engine rebuilds, the number of development and certification engine firings and the number of Flight Test engines. Once these items were defined a typical WBS structure was generated for each development phase. The same general WBS structure was used for all of the baseline engines but it was tailored as necessary to account for engine cycle differences. The WBS for each development program was then broken into separate task activities applicable to the particular engine being estimated and costs were estimated for each task and activity. Once these estimates were complete the individual elements were added to obtain total costs for each development phase and engine cycle. Table 10 shows a typical WBS structure used to estimate development costs for the baseline engines.

Engineering labor costs were estimated from head count estimates that were based on historical data from other programs. Hardware costs were derived from the engine production cost estimates using factors to account for the development hardware being early units on the learning curve and more fully instrumented than production engines. Government testing costs were derived from testing information available from other programs such as the STME and SMV programs. Propellant costs were estimated from propellant flow rates obtained from the engine cycle studies assuming average run times and thrust levels for the tests. Costs for engineering labor to support the tests and analyze results were estimated from head count estimates for these tasks.

This approach resulted in detailed development cost estimates being generated for the baseline engines. Having baseline estimates available at the individual task and activity level provided insight into the development cost drivers and it made the evaluation of new technologies a simpler task.

Development Cost Estimates Results

Development cost estimates were another factor in making the down select from six to three engines during the TA-3 and TA-4 studies. As with production costs, absolute development costs for the baseline engines have not been included in this report because of the sensitivity of these costs to vehicle/architecture specific and capture model assumptions. Again, a generic set of assumptions was used to calculate development costs, and then normalized, with the data reported as a delta from a baseline. Again, this does not provide absolute development cost data (which is inaccurate unless tied to a specific architecture and capture model), but rather, a basis to compare the projected development costs of the different cycles.

In the technology assessment portion of this report, the cost impacts of the new technologies are shown as delta development costs relative to these baseline engine development costs.

Table 10 Typical Work Breakdown Structure Used for Engine Development Cost Estimates

1	Engine Definition and FSD Design
1.1	Program Management
1.2	Development Engineering Management
1.3	System Engineering and Integration
1.3.1	Engine Systems Analysis and Integration
1.3.2	Engine System Design and Component Integration
1.3.3	Reliability and Mission Assurance
1.3.4	Quality Assurance
1.3.5	System Cost Analysis
1.3.6	Configuration Management
1.4	Engine Component Development
1.4.1	Engine Thrust Chamber and Nozzle Assembly (TCA)
1.4.1.1	Engine Definition
1.4.1.2	Preliminary Design
1.4.1.3	Final Design
1.4.2	Engine Turbopumps
1.4.2.1	Engine Definition
1.4.2.2	Preliminary Design
1.4.2.3	Final Design
1.4.3	Engine Preburners, Hot Gas Systems and Gox Hex
1.4.3.1	Engine Definition
1.4.3.2	Preliminary Design
1.4.3.3	Final Design
1.4.4	Engine Controls
1.4.4.1	Engine Definition
1.4.4.2	Preliminary Design
1.4.4.3	Final Design
1.4.5	Engine Externals (Ducting and Support Devices)
1.4.5.1	Engine Definition
1.4.5.2	Preliminary Design
1.4.5.3	Final Design
1.4.6	Logistics Support Planning
1.4.7	Travel
2	Full Scale Development
2.1	Program Management
2.2	Development Engineering Management
2.3	System Engineering and Integration
2.3.1	Engine System Analysis and Integration
2.3.2	Engine System Design and Component Integration
2.3.3	Reliability and Mission Assurance
2.3.4	Quality Assurance
2.3.5	System Cost Analysis
2.3.6	Configuration Management
2.4	Engine Component Development
2.4.1	Engine Thrust Chamber and Nozzle Assembly (TCA)
2.4.1.1	Development Engineering Support
2.4.1.2	Tooling and STE
2.4.1.3	New Development Hardware
2.4.1.4	Development Engine Rebuilds
2.4.1.5	Certification Engine Hardware
2.4.1.6	DVS and Component Testing
2.4.1.7	Powerhead and Engine Test Support
2.4.2	Engine Turbopumps
2.4.2.1	Development Engineering Support
2.4.2.2	Tooling and STE
2.4.2.3	New Development Hardware
2.4.2.4	Development Engine Rebuilds
2.4.2.5	Certification Engine Hardware
2.4.2.6	DVS Testing
2.4.2.7	Boost Pump Testing
2.4.2.8	High Pressure Turbopump Testing
2.4.2.9	Powerhead and Engine Test Support
2.4.3	Engine Preburners, Hot Gas Systems and Gox Hex
2.4.3.1	Development Engineering Support
2.4.3.2	Tooling and STE
2.4.3.3	New Development Hardware
2.4.3.4	Development Engine Rebuilds
2.4.3.5	Certification Engine Hardware
2.4.3.6	DVS Testing
2.4.3.7	Preburner Testing
2.4.3.8	Turbopump, Powerhead and Engine Test Support
2.4.4	Engine Controls
2.4.4.1	Development Engineering and Supplier Support
2.4.4.2	Tooling and STE

2	Full Scale Development (continued)
2.4.5.4	Development Engine Rebuilds
2.4.5.5	Certification Engine Hardware
2.4.5.6	DVS and Component Testing
2.4.5.7	Powerhead and Engine Testing Support
2.5	Engine Assembly
2.5.1	Planning
2.5.2	Tooling and STE
2.5.3	New Development Engine Assembly
2.5.4	Teardown and Rebuild Assembly
2.5.5	Certification Engine Assembly
2.6	Powerhead Testing
2.6.1	Planning
2.6.2	Test STE
2.6.3	Testing Costs (Govt Costs)
2.6.4	Contractor Test Support
2.6.5	Propellants
2.7	Development Engine Testing
2.7.1	Planning
2.7.2	Test STE
2.7.3	Testing Costs (Govt Costs)
2.7.4	Contractor Test Support
2.7.5	Propellants
2.8	Environmental Engine Testing
2.8.1	Planning
2.8.2	Testing Costs (Subcontractor Costs)
2.8.3	Contractor Test Support
2.9	Certification Engine Testing
2.9.1	Planning
2.9.2	Testing Costs (Govt Costs)
2.9.3	Contractor Test Support
2.9.4	Propellants
2.10	Logistical Support
2.10.1	Analysis and Planning
2.10.2	Training
2.10.3	Technical Data and Manuals
2.10.4	Overhaul and Repair Planning
2.10.5	Ground Support Equipment
2.10.6	Shipping Containers
2.11	Facility Modifications and Equipment
2.12	Travel
3	Flight Test Support
3.1	Flight Test Engines
3.1.1	Hardware and Assembly
3.1.2	Acceptance Testing
3.2	Flight Test Engine Support
3.2.1	System Engineering and Integration Support
3.2.1.1	Systems Integration and Analysis
3.2.1.2	Systems Analysis
3.2.1.3	Reliability and Mission Assurance
3.2.2	Engine System and Component Support
3.2.2.1	Contractor On-site Engine Support
3.2.2.2	Contractor In-house Engine Support
3.3	Travel
4	Life Extension Program
4.1	Program Management
4.2	Development Engineering Management
4.3	System Engineering and Integration Support
4.3.1	Engine System Analysis and Integration
4.3.2	Engine System Design and Component Integration
4.3.3	Reliability and Mission Assurance
4.3.4	Quality Assurance
4.3.5	System Cost Analysis
4.3.6	Configuration Management
4.4	Engine Component Development
4.4.1	Engine TCA Support
4.4.2	Engine Turbopump Support
4.4.3	Engine Preburner, Hot Gas and Gox Hex Support
4.4.4	Engine Controls Support
4.4.5	Engine Externals Support
4.5	Engine Overhaul
4.5.1	TCA Rebuilds
4.5.2	Turbopump Rebuilds

Table 11 compares development costs for the three baseline 600K staged combustion engines and the two 250K engines. The development costs show similar trends to the engine production costs with the DBFFSC dual preburner full flow staged combustion cycle engine having the highest development cost and the SBFRSC single preburner fuel rich staged combustion engine having the lowest development cost of the three staged combustion engines. Hardware cost differences account for most of this development cost difference; however, there is less component testing with the single preburner which also reduces development costs for that engine.

Table 11 Comparison of Development Cost for Baseline Engines

	Engine Cycle Configuration	Relative Development Cost
600K Thrust Engines		
DBFRSC	Staged Combustion Cycle with Dual Fuel Rich Preburners	Base
DBFFSC	Staged Combustion Cycle with Dual Full Flow Preburners	102.0%
SBFRSC	Staged Combustion Cycle with Single Fuel Rich Preburner	96.6%
250K Thrust Engines		
SBFRGG	Gas Generator Cycle with Single Fuel Rich Gas Generator	Base
SPLTEX	Split Expander Cycle	96.1%

Note: 250K SBFRGG Cycle Does Not Have Boost Pumps

For the 250K engines the SPLTEX expander cycle engine has a slightly lower development cost than the SBFRGG gas generator cycle engine. This difference is primarily because of hardware cost differences. The expander cycle engine has no gas generator component tests but it does have boost pump tests that the gas generator engine does not have. As a result the total development costs for both engines are close.

5.3.5 Baseline Engine Operations Cost Estimates

Operations cost estimates were also made for each of the baseline engines as part of the TA-3 engine studies. This section discusses the methodology used and results obtained for the baseline operations cost estimates.

Operations Cost Methodology

Operations cost estimates were made for each of the six baseline engines. Operational assumptions used for the operations cost calculations were taken directly from the study costing ground rules and assumptions (See Section 5.2). Scheduled and unscheduled engine maintenance, sustaining engineering, anomaly resolution and replacement engine costs were included in the operations cost estimates. The costs calculated were representative of average costs over the 20 year operational flight period. Costs per engine per mission, annual costs and total costs over the 20 year period were determined for each of the engines. The operations costs were calculated using an Excel spreadsheet program that book kept all of the operations cost elements. A flow chart showing the operations cost methodology used is shown in Figure 23.

To calculate scheduled maintenance costs, turnaround tasks required for each of the baseline engines were identified and the man-hours required to complete each task were estimated. Periodic inspections that might be required for the engines were identified and the man-hours included for them. Costs were then calculated for all of these scheduled events taking into account how frequent the tasks are performed, the man hours required to complete them and labor costs per man-hour. The other scheduled maintenance activity that was addressed is engine overhauls. In the 2GRLV program, the engines are designed for a life of 100 missions with a scheduled overhaul at 50 missions.

To calculate overhaul costs for the baseline engines, the quantity of engines in the fleet expected to have overhauls was determined and the average cost of an overhaul estimated. The average overhaul costs were defined as a percentage of the hardware costs for a new engine. As a result, engine overhaul costs varied as engine production costs changed. In addition to the direct overhaul costs, the cost of acceptance testing the engines at SSC was estimated and included.

Unscheduled maintenance costs consist of unscheduled engine and LRU removal costs and the costs to repair failed components and engines. The cost of acceptance testing the repaired components and engines is also an unscheduled maintenance cost element. Maintenance rates for the baseline engine components were defined as part of the reliability studies for the baseline engines. The rates from those studies were used in the operations cost analyses to drive the component and engine removal and repair rates. To estimate unscheduled maintenance costs, the engine components were segregated into five groups. Components in the first four groups can be replaced at the launch site. The groups consisted of small LRU components such as control components that can be easily replaced, larger LRU components such as boost pumps that are more difficult to replace, the two high pressure turbopumps, the nozzle and components such as the combustion chamber that require an engine disassembly. Average component removal and replacement man-hours were established for each of the groups of components that can be replaced at the launch site. Engine removal and replacement man-hours were established for the last group of components that require an engine disassembly. Different removal and replacement man-hours, that take into account whether the replacements occur on the launch pad or in the vehicle processing facility, were established for each of the groups. Estimates were made of the percentage of removals that would occur on the pad and in the processing facility.

Average component repair costs were then estimated for each of the groups of components. As with the engine overhaul costs, the component repair costs were defined as a percentage of new engine hardware costs. For the components that could not be removed at the launch site, it was assumed that the whole engine would be returned to SSC and repaired there.

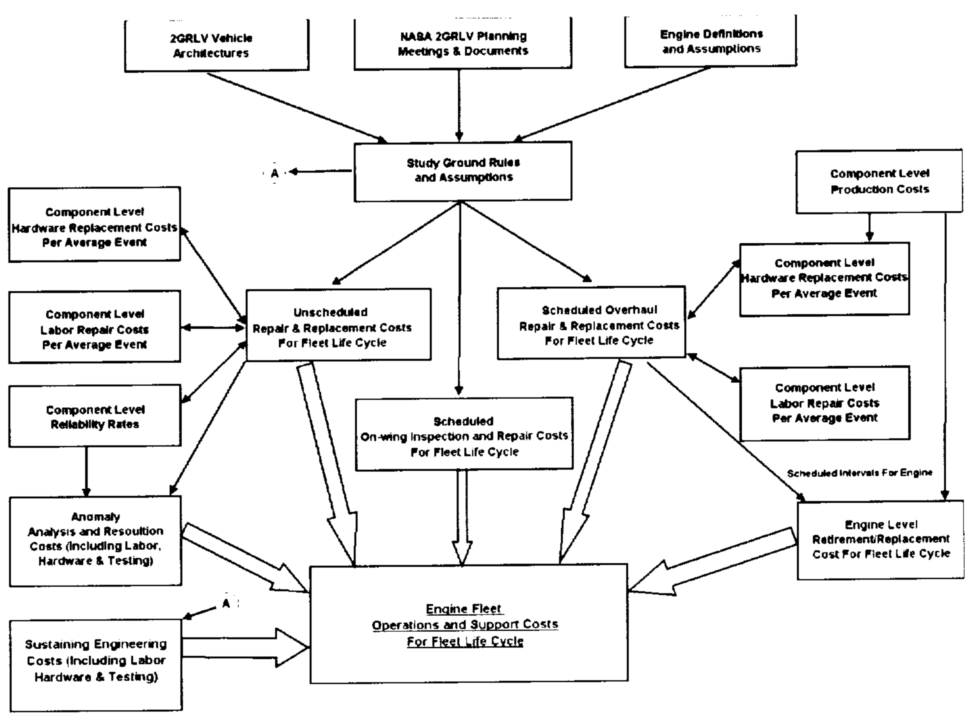


Figure 23 Flow Process for Engine Operations Cost Calculations

Acceptance tests were included in the unscheduled maintenance costs. For the components that are replaced at the launch site, it was assumed that the component acceptance tests would be piggybacked on other engines and the components would bear only a portion of the test costs. For the cases where the whole engine is returned to SSC for repair, the engine would bear the entire test cost. Propellant costs were included as part of the acceptance test costs, and packaging and shipping costs were included when appropriate.

Sustaining engineering was also included in the operations cost estimates for the baseline engines. Engineering labor costs were calculated from a head count estimate that was made using historical data from other programs. Engine hardware costs were estimated from new engine costs. The hardware costs for each year were prorated based on the number of engine tests made. Engine testing costs in support of sustaining engineering were estimated using an assumed number of tests each year and SSC testing costs from other programs.

Anomaly resolution costs were estimated by assuming a portion of the unscheduled engine component failures result in anomaly investigations. The anomalies were categorized into large and small investigations with most anomalies falling in the small category. Engineering head counts (above sustaining engineering) were estimated for each anomaly category and labor costs were calculated from these estimates. Engine hardware and testing costs for anomaly investigations were estimated in the same manner as for sustaining engineering.

Replacement engine hardware costs were also included in the baseline engine operations cost estimates. An engine attrition rate due to damage was assumed and the cost of replacing those engines was calculated from the new production engine costs. The replacement costs for installed and spare engines that are retired because of life limits were calculated on a cost per engine mission basis. The cost of a new replacement engine was amortized over its 100 mission life resulting in an engine replacement cost for each engine mission of one per cent of a new engine cost. Operations cost estimates were calculated for the baseline engines both with and without replacement engine costs included.

The approach used to calculate operations costs for the baseline engines provided detailed operations cost estimates. Calculating each cost element separately facilitated the evaluation of new technologies for the baseline engines.

Baseline Engine Operations Cost Estimate Results

As with the other costs, operations cost estimates for the baseline engines were one of the items used to down select from the six baseline cycles to the three engines carried forward for further evaluation in this study. The baseline estimates also provided a base from which to evaluate the cost impacts that new technologies would have on engine operations costs.

As with the other costs, absolute operations cost estimates for the six baseline engines have not been included in this report. They were used as base costs in the assessment of new technologies for each engine. In the technology assessment portion of this report the impacts of the new technologies on engine operations costs are shown as delta costs from the baseline operations cost estimates.

Table 12 shows relative operations costs for the three 600K staged combustion engines and for the two 250K engines. The DBFFSC dual preburner full flow staged combustion engine has the highest operations cost while the SBFRSC single preburner fuel rich staged combustion engine has the lowest operations cost of the three 600K engines. A portion of the reduction for the SBFRSC engine is due to lower unscheduled maintenance rates; however, most of it is due to lower engine overhaul and repair hardware costs.

For the two 250K engines the SPLTEX expander cycle engine has a much lower operations cost than the SBFRGG gas generator cycle engine. Unscheduled maintenance rates as well as overhaul and repair hardware costs are lower for the SPLTEX engine accounting for its significant reduction in operations costs.

Table 12 Comparison of Operations Costs for Baseline Engine Cycles

	Engine Cycle Configuration	Relative Operations Cost
600K Thrust Engines		
DBFRSC	Staged Combustion Cycle with Dual Fuel Rich Preburners	Base
DBFFSC	Staged Combustion Cycle with Dual Full Flow Preburners	100.6%
SBFRSC	Staged Combustion Cycle with Single Fuel Rich Preburner	96.6%
250K Thrust Engines		
SBFRGG	Gas Generator Cycle with Single Fuel Rich Gas Generator	Base
SPLTEX	Split Expander Cycle	93.1%

Note: (1) Operations Costs are Costs Per Engine Per Mission
(2) 250K SBFRGG Cycle Does Not Have Boost Pumps

5.3.6 Cost Analysis - Three Selected Cycles

After down select to the three selected cycles, additional cost analyses were performed for these engines as part of the TA-3 and TA-4 studies. These analyses primarily involved evaluating the cost impacts of new technologies identified as candidates for the 2GRLV engines. The following sections discuss the cost analyses performed for the three selected cycles.

5.3.7 Approach for Evaluating Cost Impacts of New Technologies

The cost impacts of incorporating new technologies were determined for each of the three selected engines. The impacts on production, development and operations costs were quantified for each technology. The same costing ground rules and assumptions used for the baseline engine estimates were used for the technology evaluations.

As discussed in Section 5.3.1-5.3.3, costs for the baseline engines were calculated at a detailed level with engine costs being determined at a component and/or activity level. This detailed approach for the baseline engine estimates made the evaluation of the new technologies a simple process. To evaluate a new technology all components and/or activities effected by the technology were identified. New estimates were then made for each of the effected items assuming the new technology was incorporated in the engine. The line items in the original estimates that changed were replaced with the new estimates and all of the cost elements summed to obtain new costs for the engine with the technology included. This process was used to evaluate the impact of new technologies on production, development and operations costs for the three selected engine cycles.

Using the above approach the cost estimates previously made for the baseline engines became the base costs for quantifying the technology impacts. The cost impacts of each technology were determined as delta costs relative to these baseline estimates. Since the baseline costs are different for each cycle, each engine has a different set of base values.

For the initial evaluation of new technologies each technology was incorporated in the engines one at a time and the cost changes determined. Each of the technologies shown in Section 4.0 was evaluated in the three engines in this manner (if applicable to the engine). For some technologies there are interactions with other technologies that can affect the magnitudes of the cost impacts that occur. To quantify the effects of combined technologies additional assessments were made with all of the technologies in the engines at the same time. For these evaluations applicable interactions were taken into account and included. The cost deltas obtained from these analyses are slightly different than if the individual technology cost deltas are added together. The results of both assessments are included in the final report.

A total of 17 new technologies were evaluated as candidates for the 2GRLV engines. Not all were applicable to all of the selected engines. The cost results obtained are discussed in the following sections.

Impacts of New Technologies on Engine Production Costs

Table 13 shows the impacts that each new technology has on engine production costs for the three selected cycles. The initial costs presented in this table are for each technology incorporated separately. The cost deltas shown are the changes that occur in engine production cost relative to the baseline cost due to incorporating the new technology in the engine. Cost deltas are summed at the bottom of the table for all of the technologies.

The Controller with Integral EHMS has the largest impact on engine production cost. The impact is similar for all three engines. The baseline engine uses a control system with hydraulic actuators while the incorporation of the Controller with Integral EHMS includes a fully integrated electronic control system in conjunction with the EHMS. This accounts for the large cost impact of this system. The next largest impact on production cost is the incorporation of electro-mechanical actuators for the control valve and TVC actuators. The EMA control valve actuators are also included in the Controller with Integral EHMS cost impacts.

The largest reductions in production cost occur with the milled channel nozzle and the advanced hydrogen turbopump. Both offer the potential for significant savings in engine production cost. Cost impacts are included for the two containment systems being considered for the engines. These are very rough estimates since design configurations are not available for either of these systems at this time.

Production cost impacts for each technology individually, as well as combined, are shown in Table 13 for each of the three selected engine cycles. The largest increase is for the 1000K SBORSC engine cycle, while the smallest increase is for the 600K SBFRSC engine. Because of interactions, the total combined cost impacts are lower than the sum of the individual cost impacts. The technologies that have interacting effects are:

1. The milled channel nozzle and the LOX cooled nozzle section replacing the oxidizer heat exchanger,
2. The improved durability MCC and the use of split circuit cooling for the MCC,
3. The improved durability main injector and the improved combustion efficiency, and
4. The use of EMA actuators with a fully electronic control system and the incorporation of an advanced controller with integral EHMS.

	Each Technology Evaluated Separately Relative to Baseline	Δ Production Cost, K$ SBFRSC	Δ Production Cost, K$ SPLTEX	Δ Production Cost, K$ SBORSC
Baseline Engine Cost		Base	Base	Base
Technology (Delta Cost Relative to Baseline Engine Cost)				
1	Milled Channel Nozzle	-$622	-$466	-$818
2	Fail Safe Hot Gas System	$155	N/A	N/A
3	Improved Durability MCC	-$389	-$350	-$450
4	Improved Durability Preburner Injectors	-$39	N/A	-$41
5	Improved Durability Main Injectors	-$86	-$78	-$98
6	LOX Cooled Nozzle Section Replacing Gox Heat Exchanger	$233	$194	N/A
7	Electromechanical/Electro-Pneumatic Actuators/Assoc Controller	$626	$443	$699
8	Avanced High Speed Main Hydrogen Pump (Unshrouded Impellers & Hydrostatic Bearings)	-$933	-$466	N/A
9	Controller with Integral EHMS	$1,983	$1,854	$2,025
10	External Gas Containment System	$295	$249	$368
11	External Material Containment System	$401	$350	$532
12	Increased Main LOX Pump Efficiency	$117	$78	$164
13	Increased Main Kerosene Pump Efficiency	N/A	N/A	$123
14	Hydraulic Fuel Boost Turbine	N/A	-$117	N/A
15	Low Oxidizer Inlet Pressure Capability	$47	$31	$57
16	Split Circuit MCC Cooling	N/A	$280	N/A
17	Increased Combustion Efficiency (Main Injector Improvement)	$124	$117	$147
	Sum of Delta Costs for Individual Technologies	$1,914	$2,118	$2,707
	All Technologies Together in Engine (Includes Interactions)	$1,587	$1,840	$2,466
Note: All Costs are in Thousands of FY2000 Dollars				

Table 13 Technology Impacts on Engine Production Costs

Impacts of New Technologies on Engine Development Cost

The impacts that each technology has on engine development costs are shown in Table 14. The initial cost deltas shown are for each technology incorporated separately. As with the production costs these cost deltas are the changes that occur in engine development costs relative to the baseline engine costs as each technology is incorporated. Cost deltas for all of the technologies are summed at the bottom the table.

The Controller with Integral EHMS has the largest impact on engine development cost. With this system there are both significant engineering and hardware cost impacts. As discussed earlier the EHMS includes a fully integrated electronic control system so there is significant development effort involved in incorporating this on the engine. The next highest impact is the incorporation of EMA's on the engine. The control valve EMA's are also part of the Controller with Integral EHMS and they are included in those development cost deltas as well.

The technology items causing the largest reduction in engine development costs are the milled channel nozzle and the advanced hydrogen turbopump. These reductions occur because of lower hardware costs for the engine development program with these technologies.

Cost impacts are shown for developing the two containment systems as part of the engine. These are very rough estimates since the configurations for these two systems are not defined at this time.

Development cost impacts for each of the technologies individually, as well as combined, are shown in Table 14 for each of the three selected cycles. As with the combined production cost impacts, the cost impacts for all technologies combined is different than the sum of the individual impacts, due to the interactions discussed previously.

Table 14 Technology Impacts on Engine Development Costs

	Each Technology Evaluated Separately Relative to Baseline	Δ Development Cost, M$ SBFRSC	Δ Development Cost, M$ SPLTEX	Δ Development Cost, M$ SBORSC
Baseline Engine Cost		Base	Base	Base
Technology (Delta Cost Relative to Baseline Engine Cost)				
1	Milled Channel Nozzle	-$24.5	-$18.5	-$30.4
2	Fail Safe Hot Gas System	$6.2	N/A	N/A
3	Improved Durability MCC	-$15.3	-$13.6	-$16.2
4	Improved Durability Preburner Injectors	-$1.3	N/A	-$1.2
5	Improved Durability Main Injectors	-$3.1	-$2.8	-$3.3
6	LOX Cooled Nozzle Section Replacing Gox Heat Exchanger	$12.8	$10.3	N/A
7	Electromechanical/Electro-Pneumatic Actuators/Assoc Controller	$21.8	$15.7	$23.3
8	Avanced High Speed Main Hydrogen Pump (Unshrouded Impellers & Hydrostatic Bearings)	-$25.7	-$9.4	N/A
9	Controller with Integral EHMS	$105.1	$101.2	$103.9
10	External Gas Containment System	$14.4	$12.8	$16.2
11	External Material Containment System	$16.2	$14.5	$19.1
12	Increased Main LOX Pump Efficiency	$8.0	$6.2	$9.2
13	Increased Main Kerosene Pump Efficiency	N/A	N/A	$7.6
14	Hydraulic Fuel Boost Turbine	N/A	$0.1	N/A
15	Low Oxidizer Inlet Pressure Capability	$6.9	$6.1	$7.4
16	Split Circuit MCC Cooling	N/A	$12.1	N/A
17	Increased Combustion Efficiency (Main Injector Improvement)	$4.3	$4.4	$4.7
	Sum of Delta Costs for Individual Technologies	$125.8	$139.1	$140.3
	All Technologies Together in Engine (Includes Interactions)	$113.9	$129.1	$131.5

Note: All Costs are in Millions of FY2000 Dollars

Impacts of New Technologies on Engine Operations Cost

Operations cost impacts for each of the new technologies are shown in Table 15. Again, the cost impacts are shown for each technology individually, as well as combined. Again, the sum of the individual impacts does not equal the combined impact due to interactions discussed previously.

Significant operations cost savings occur with the milled channel nozzle, the improved durability MCC, and the advanced high pressure hydrogen turbopump. These savings primarily result from lower engine hardware costs, improved reliability and reduced sustaining engineering requirements with these technologies.

The Controller with Integral EHMS also provides significant operations cost savings for the engines. These savings are primarily due to lower scheduled turnaround and inspection costs and reduced fault isolation costs with the EHMS. Electro-mechanical actuators (EMA's) reduce operations cost but their savings are not as great as for the other technologies. EMA's simplify engine checkouts and they permit the hydraulic system to be eliminated from the engine, but a large part of their savings are offset by higher actuator hardware costs.

None of the technologies cause significant operations cost increases but small increases occur for several of the technologies. The largest increases are for the engine containment systems. These increases are primarily due to higher engine hardware costs with the containment systems included. The operations cost estimates for these two technologies are rough estimates since design configurations for the two containment systems are not defined at this time.

All of the engine cycles show significant savings in operations costs with all technologies incorporated. The largest savings are for the 600K SBFRSC cycle engine, while the smallest savings are for the 250K SPLTEX engine.

Table 15 Technology Impacts on Engine Operations Costs

	Each Technology Evaluated Separately Relative to Baseline	Δ Operations Cost K$/Eng/Mission SBFRSC	Δ Operations Cost K$/Eng/Mission SPLTEX	Δ Operations Cost K$/Eng/Mission SBORSC
	Baseline Engine Cost	Base	Base	Base
	Technology (Delta Cost Relative to Baseline Engine Cost)			
1	Milled Channel Nozzle	-$140.0	-$116.9	-$194.2
2	Fail Safe Hot Gas System	$4.7	N/A	N/A
3	Improved Durability MCC	-$140.9	-$105.2	-$182.6
4	Improved Durability Preburner Injectors	-$19.1	-$8.8	-$23.0
5	Improved Durability Main Injectors	-$22.5	-$19.2	-$30.6
6	LOX Cooled Nozzle Section Replacing Gox Heat Exchanger	$5.6	$3.8	N/A
7	Electromechanical or Electro-Pneumatic Actuators	-$28.8	-$29.8	-$33.8
8	Advanced High Speed Main Hydrogen Pump (Unshrouded Impellers & Hydrostatic Bearings)	-$141.1	-$94.3	N/A
9	Controller with Integral EHMS	-$93.9	-$82.0	-$107.8
10	External Gas Containment System	$9.0	$5.0	$13.7
11	External Material Containment System	$12.2	$10.5	$19.9
12	Increased Main LOX Pump Efficiency	$4.6	$2.9	$7.3
13	Increased Main Kerosene Pump Efficiency	N/A	N/A	$5.5
14	Hydraulic Fuel Boost Turbine	N/A	-$3.5	N/A
15	Low Oxidizer Inlet Pressure Capability	$1.4	$0.4	$2.1
16	Split Circuit MCC Cooling	N/A	$8.9	N/A
17	Increased Combustion Efficiency (Main Injector Improvement)	$4.0	$3.7	$5.8
	Sum of Delta Costs for Individual Technologies	-$544.8	-$424.5	-$517.6
	All Technologies Together in Engine (Includes Interactions)	-$527.2	-$401.8	-$635.7

Note: (1) Costs Shown are Costs Per Engine Per Mission in Thousands of FY2000 Dollars
(2) Includes Cost of Replacement Engines for Retired Engines

5.4 STUDY SUMMARY

The trade studies described in the sections above are summarized in Tables 16-18. For the performance and reliability columns, positive numbers are a benefit. For the cost columns positive indicates a cost increase so negative numbers in the cost columns are a benefit. Therefore positive cost numbers are shown in red.

These tables were generated so that reliability vs. cost vs. performance comparisons could easily be performed. This allows first order summaries of combinations of technologies to be generated to aid in deciding the final propulsion system configurations to match vehicle requirements. There is some inter-connectivity between the technologies as can be seen in the summary at the bottom of each table. The summary line is the result of a study that included all of the appropriate technologies for each cycle. The values in this summary line do not equal the simple addition of the column entries above. Therefore, when final engine selection is being made, a final run of the models with the desired technology combinations should be done.

The LOV values in the table reflect the reliability analysis conducted with median levels of expected effectiveness for improvements generated by the new technologies. The resultant numbers showed significant improvement over the baseline (SSME). Based on the space Shuttle Quantitative Risk Assessment System (QRAS) data, the loss of vehicle (LOV) rate due to main engine is currently 258 per million missions. Depending upon the assumptions of effectiveness and implementation success of the technologies, the resultant safety of the engines ranges from 5 to 45 LOV events per million.

Table 16 SBFRSC Cycle Technology Improvement Parametric Results

SBFRSC Cycle Technology Improvement Parametric Results							
	TECHNOLOGY	Delta LOV (per million)	Delta ISP-Vac (Sec)	Delta ISP-S/L (Sec)	Delta DDT&E Cost ($M)	Delta Production Cost ($K)	Delta O&S Cost ($K/eng/mission)
1	Improved Durability Combustion Chamber	24.8	0	0	-$15.3	-$389.0	-$140.9
2	Fail Safe Hot Gas System	20.93	---	---	$6.2	$155.0	$4.7
3	Milled Channel Nozzle	15.84	0	0	-$24.5	-$622.0	-$140.0
4	High Speed Main Fuel Pump	13.14	0	0	-$25.7	-$933.0	-$141.1
5	Improved Durability Injectors	7.58	---	---	-$4.4	-$125.0	-$41.6
6	LOX Cooled Nozzle Section	0.97	0	0	$12.8	$233.0	$5.6
7	Electromechanical / Electro-pneumatic Actuators	3.55	---	---	$21.8	$626.0	-$28.8
8	Controller w/ Integrated EHMS	41.13	---	---	$105.1	$1,983.0	-$93.9
9	External Material Containment System	16.66	---	---	$16.2	$401.0	$12.2
10	External Gas Containment System	16.56	---	---	$14.4	$295.0	$9.0
11	Hydraulic Fuel Boost Turbine	---	---	---	---	---	---
12	Low Oxidizer Inlet Pressure	---	0	0	$6.9	$47.0	$1.4
13	Split Circuit Cooling	---	---	---	---	---	---
14	Increased Main Fuel Pump Efficiency	---	---	---	---	---	---
15	Increased Main Oxidizer Pump Efficiency	---	0	0	$8.0	$117.0	$4.6
16	Increased Combustion Efficiency	---	0.9	0.9	$4.3	$124.0	$4.0
	Summary Package with All Technologies	161.16	0.91	0.91	$113.9	$1,587.0	-$527.2

Table 17 SPLTEX Cycle Technology Improvement Parametric Results

SPLTEX Cycle Technology Improvement Parametric Results							
	TECHNOLOGY	Delta LOV	Delta ISP-Vac (Sec)	Delta ISP-S/L (Sec)	Delta DDT&E Cost ($M)	Delta Production Cost ($K)	Delta O&S Cost ($K/eng/mission)
1	Improved Durability Combustion Chamber	19.73	0	0	-$13.6	-$350.0	-$105.2
2	Fail Safe Hot Gas System	---	---	---	---	---	---
3	Milled Channel Nozzle	16.37	0	0	-$18.5	-$466.0	-$116.9
4	High Speed Main Fuel Pump	7.05	0	0	-$9.4	-$466.0	-$94.3
5	Improved Durability Injectors	7.58	0.88	0.88	-$2.8	-$78.0	-$28.0
6	LOX Cooled Nozzle Section	0	0	0	$10.3	$194.0	$3.8
7	Electromechanical / Electro-pneumatic Actuators	3.48	---	---	$15.7	$443.0	-$29.8
8	Controller w/ Integrated EHMS	37.83	---	---	$101.2	$1,854.0	-$82.0
9	External Material Containment System	14.78	---	---	$14.5	$350.0	$10.5
10	External Gas Containment System	14.77	---	---	$12.8	$249.0	$5.0
11	Hydraulic Fuel Boost Turbine	---	2.8	2.8	$0.1	-$117.0	-$3.5
12	Low Oxidizer Inlet Pressure	---	0	0	$6.1	$31.0	$0.4
13	Split Circuit Cooling	---	0	0	$12.1	$280.0	$7.7
14	Increased Main Fuel Pump Efficiency	---	---	---	---	---	---
15	Increased Main Oxidizer Pump Efficiency	---	0	0	$6.2	$78.0	$2.9
16	Increased Combustion Efficiency	---	0.88	0.88	$4.4	$117.0	$3.7
	Summary Package with All Technologies	121.59	3.9	3.9	$129.1	$1,910.0	-$401.8

Table 18 SBORSC Cycle Technology Improvement Parametric Results

	SBORSC Cycle Technology Improvement Parametric Results						
	TECHNOLOGY	Delta LOV	Delta ISP-Vac (Sec)	Delta ISP-S/L (Sec)	Delta DDT&E Cost ($M)	Delta Production Cost ($K)	Delta O&S Cost ($K/eng/mission)
1	Improved Durability Combustion Chamber	27.98	0	0	-$16.2	-$450.0	-$182.6
2	Fail Safe Hot Gas System	---	---	---	---	---	---
3	Milled Channel Nozzle	18.07	0	0	-$30.4	-$818.0	-$194.2
4	High Speed Main Fuel Pump	---	---	---	---	---	---
5	Improved Durability Injectors	0	---	---	-$4.5	-$139.0	-$53.6
6	LOX Cooled Nozzle Section	---	---	---	---	---	---
7	Electromechanical / Electro-pneumatic Actuators	3.55	---	---	$23.3	$699.0	-$33.8
8	Controller w/ Integrated EHMS	34.67	---	---	$103.9	$2,025.0	-$107.8
9	External Material Containment System	16.96	---	---	$19.1	$532.0	$19.9
10	External Gas Containment System	24.65	---	---	$16.2	$368.0	$13.7
11	Hydraulic Fuel Boost Turbine	---	---	---	---	---	---
12	Low Oxidizer Inlet Pressure	---	0	0	$7.4	$57.0	$2.1
13	Split Circuit Cooling	---	---	---	---	---	---
14	Increased Main Fuel Pump Efficiency	---	0	0	$7.6	$123.0	$5.0
15	Increased Main Oxidizer Pump Efficiency	---	0	0	$9.2	$164.0	$7.3
16	Increased Combustion Efficiency	---	1.8	1.8	$4.7	$147.0	$5.8
	Summary Package with All Technologies	125.88	1.8	1.8	$131.5	$2,466.0	-$635.7

5.5 RISK REDUCTION PLAN

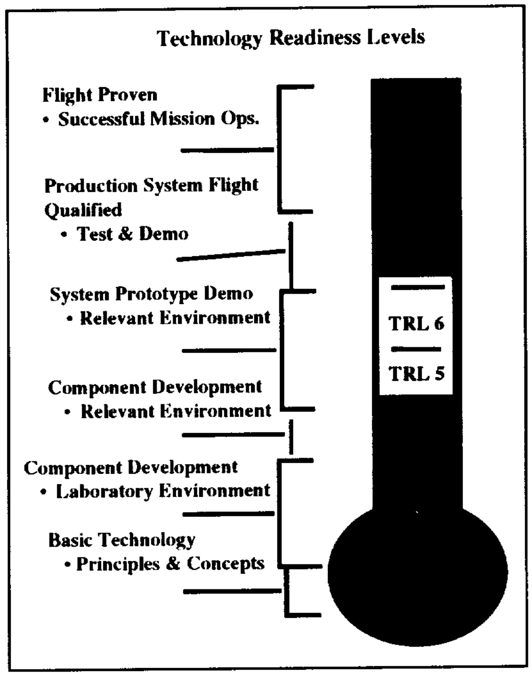
The new technologies that were added to the three selected cycles require development to bring them to the technology readiness level (TRL) required for incorporation into production rocket engines. The technologies along with their current TRL's are shown in Table 19 below. This section describes the process by which these technologies may be matured to TRL=6.

	TECHNOLOGY	TRL	SBFRSC	SPLTEX	SBORSC
1	Improved Durability Combustion Chamber	3	X	X	X
2	Fail Safe Hot Gas System	4	X		
3	Milled Channel Nozzle	4	X	X	X
4	High Speed Main Fuel Pump	3	X	X	
5	Improved Durability Main Injector Improved Durability Preburner Injector	4 3	X X	X	X X
6	LOX Cooled Nozzle Section	3	X	X	
7	Electromechanical / Electro-pneumatic Actuators	4	X	X	X
8	Controller w/ Integrated EHMS	4	X	X	X
9	External Material Containment System	3	X	X	X
10	External Gas Containment System	3	X	X	X
11	Hydraulic Fuel Boost Turbine	4		X	
12	Low Oxidizer Inlet Pressure	4	X	X	X
13	Split Circuit Cooling	4		X	
14	Increased Main Pump Efficiencies (Both Fuel & Oxidizer)	4	X	X	X
15	Increased Combustion Efficiency	4	X	X	X

Table 19 Technologies for the Configuration of improved cycles

The first ten technologies in the list are reliability improvements and the last five are performance enhancements. Risk reduction plans were constructed for these technologies by the appropriate component development team. The groups assessed the current technology levels for each technology using the guidelines shown in Figure 24 and defined the analytical work and testing required to mature the technologies to TRL=6. In some cases the plans include steps to upgrade analysis and modeling tools in order to produce the mature technologies.

Figure 24 Technology Readiness Level



Separate risk reduction plans were not created for all technologies. Split circuit cooling refers to parallel manifolding of the combustion chamber cooling passages to reduce coolant pressure drop and is therefore covered in the combustion chamber durability reduction plan. The component efficiency items refer to the normal incremental progression in component level efficiencies expected in new designs and are covered in their respective component design and development efforts.

5.5.1 Risk Reduction Plan Schedule

A notional design and development schedule was created for the five year period allotted to advance the new technologies to TRL=6 in preparation for full scale development (FSD). The schedule is shown in Figure 25 below. For the purposes of these risk reduction plans the same schedule was used for all three selected cycles. As such some of the testing activities shown at the top are not applicable for all three cycles. For example the preburner testing is not necessary for the SPLTEX cycle. At the time that the real risk reduction activities are undertaken, schedules can be optimized for each of the three cycles to minimize cost and time.

In Figure 25, a series of test programs are arranged across the top of the schedule. These tests provide an important means to advance the TRL's of the enhancing technologies. Testing begins with subscale component and rig testing to provide early design system and analytical modeling confirmation before commitment is made and final designs are undertaken. Later tests are arranged to be additive. The powerhead test includes the preburner, hot gas manifold, and turbopumps. The thrust chamber assembly (TCA) test consists of the combustion chamber, main injector, LOX cooled nozzle section and milled channel nozzle. Once these separate tests are successfully completed, the next step is to combine the two assemblies as the core engine test. Successful completion of this test leads to the addition of the boost pumps and remaining hardware to form the prototype engine test. This stepwise process insures that the components can be fully characterized in relative isolation without the system level interactivities that are present in a full engine test. Thus insuring true maturation to TRL=6 for each of the enhancing technologies.

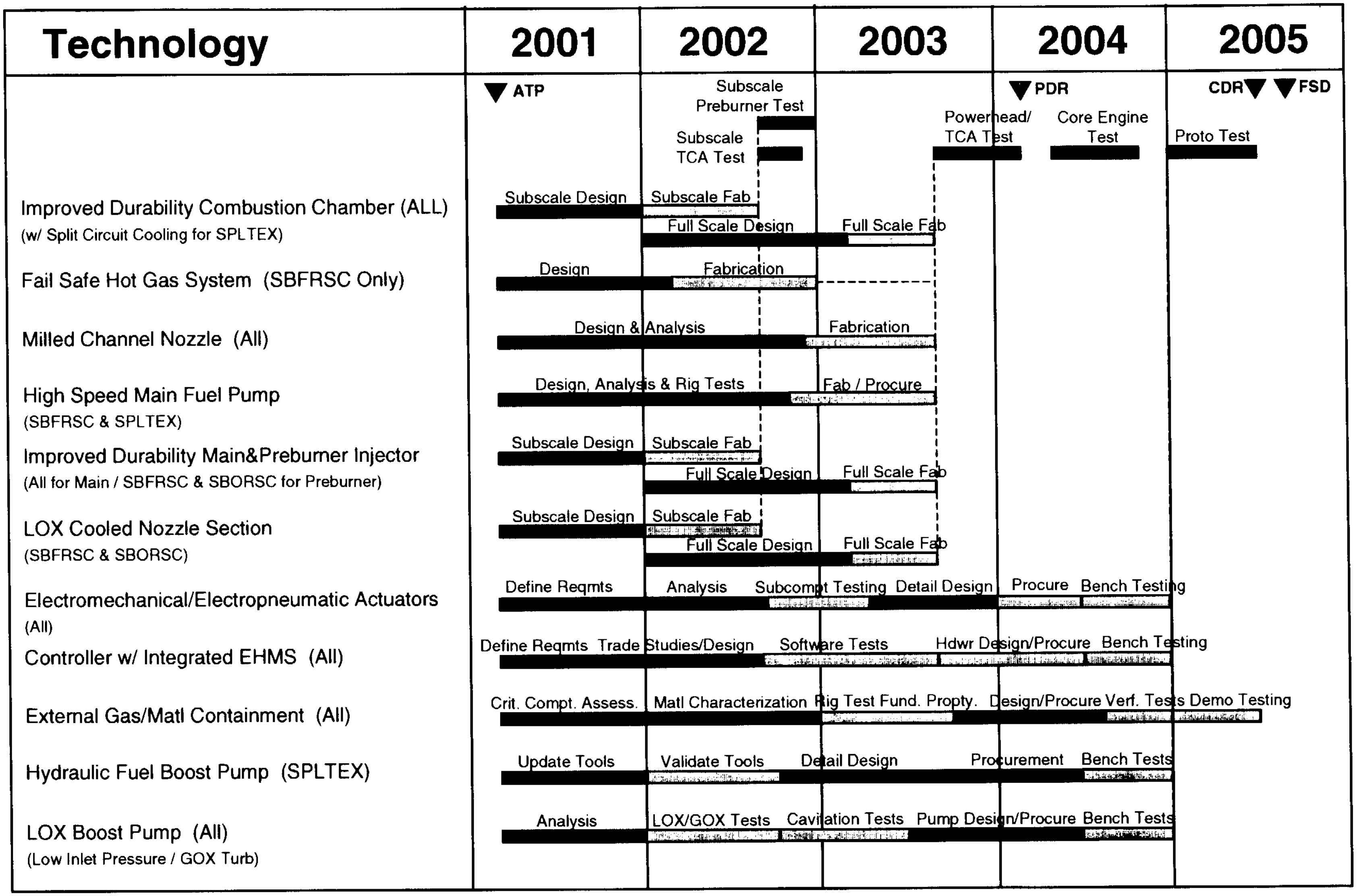


Figure 25: Notional 5-Year Technology Development Schedule

5.5.1.1 Improved Durability Combustion Chamber

As shown in the reliability analysis section the baseline combustion chamber has a large number of failure modes associated with it. The design concept chosen for this study is the formed platelet liner, which is a mature technology that has been demonstrated in the Advanced Main Combustion Chamber (AMCC) test program at MSFC. This concept allows a thinner hot gas wall with integral blanch shielding. The main risks for this design are life and producibility. Experience has shown that blanching can significantly reduce chamber life. The addition of an integral blanch shield will greatly extend chamber life. Also the thin hot gas wall will reduce wall temperatures and thermal strains. Both of which will enhance chamber life. The waterfall chart in Figure 26 and the five year schedule in Figure 25 show the details of the risk reduction plan.

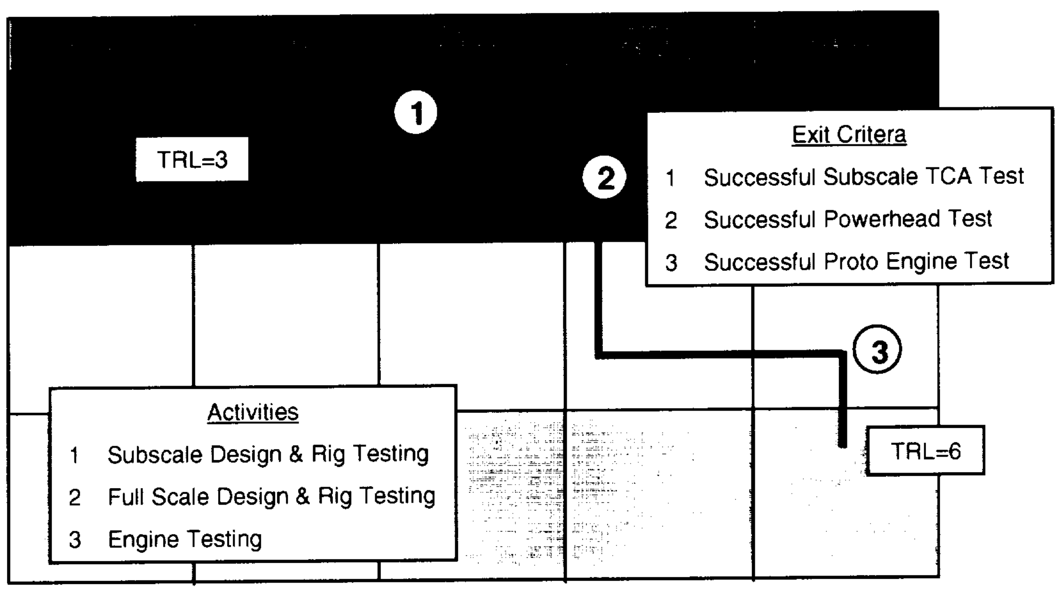


Figure 26 Improved Durability Combustion Chamber Risk Reduction Waterfall Chart

In addition to enhancing the combustion chamber durability, split circuit cooling will be integrated into the combustion chamber design to lower cooling passage pressure drop. Risks associated with meeting pressure drop goals and added complexity due to addition of coolant manifolding will be mitigated through successful design and testing. Cooling passage pressure drop reduction will be verified in the powerhead testing in 2004.

5.5.1.2 Fail Safe Hot Gas System

The intent of this technology is to minimize hot gas manifold reliability issues. The design is a double wall configuration, where the preburner to turbopump manifold is inside the turbopump to main injector manifold.

System safety is enhanced since any leak from the hot preburner manifold is contained in the cooler turbopump exhaust manifold. Leakage would be self-limiting. The turbopumps would decelerate together as a result of the leak thus maintaining mixture ratio control while the leak would be contained in the outer manifold minimizing further damage. This concept has been demonstrated previously on the P&W XLR129. The configuration described is only applicable to the SBFRSC cycle. The SPLTEX cycle does not use a preburner and the SBORSC cycle geometry has both turbopumps on a common shaft.

Risk reduction is focused on design and fabrication as shown in the waterfall chart in Figure 27 and the five-year schedule, Figure 25. The hot gas system starts at TRL = 4 which progresses to TRL = 5 with successful design and fabrication and reaches TRL = 6 with successful completion of engine testing.

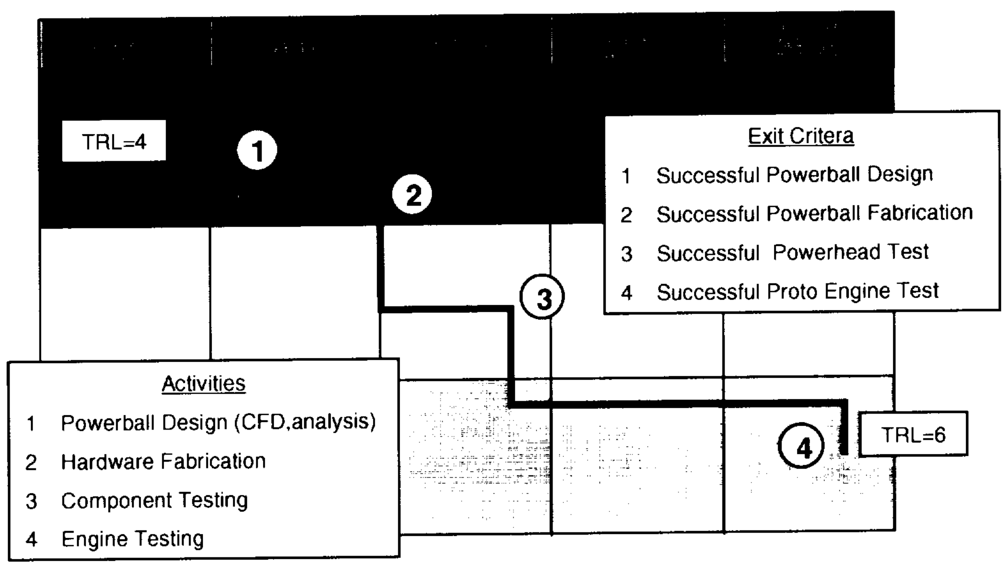


Figure 27 Fail Safe Hot Gas System Waterfall Risk Reduction Chart

5.5.1.3 Milled Channel Nozzle

The milled channel nozzle, which is applicable to all three of the selected cycles, consists of an inner wall with cooling channel milled into the outer surface. An outer wall or closeout is then attached to this inner wall completing the cooling passages. This type of construction offers more durability and simpler manufacturing and repairability than the tube type nozzles used in the baseline cycles in this study. This technology is well understood and has been used on several engines such as the RD180.

Risk reduction activity for the nozzle is focused on schedule and fabrication as shown in the waterfall chart in Figure 28 and the five-year schedule, Figure 25. The nozzle starts at TRL = 4, which advances to TRL = 5 with successful design and fabrication, and reaches TRL = 6 after successful completion of engine testing.

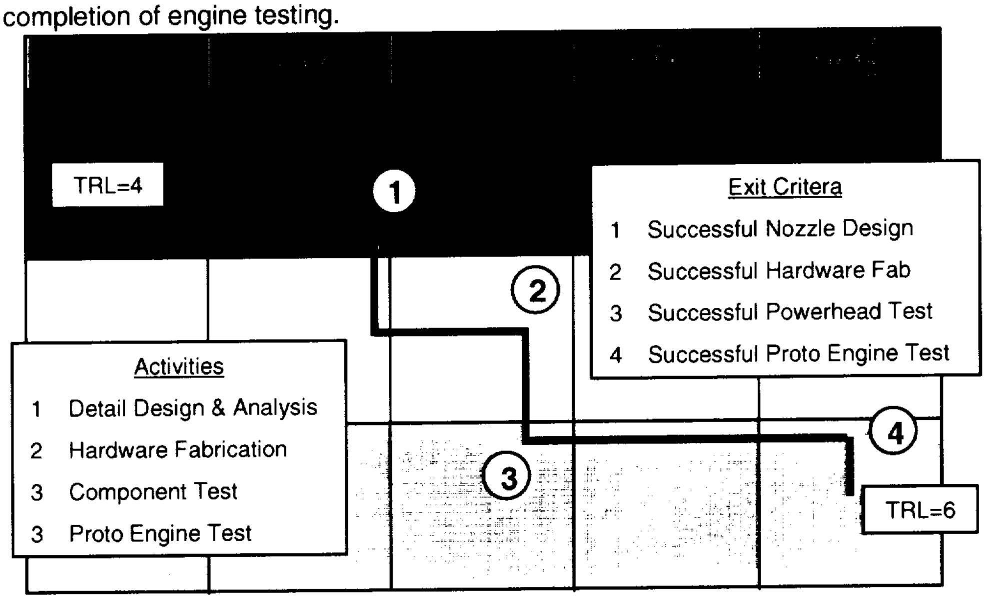


Figure 28: Milled Channel Nozzle Risk Reduction Waterfall Chart

5.5.1.4 High Speed Main Fuel Pump

By increasing the speed of the main fuel pump, a stage can be removed, resulting in lower part count and consequent reliability improvement. The high-speed fuel pump is proposed for both the SBFRSC and SPLTEX cycles. It is not applicable to the SBORSC cycle hydrocarbon fuel pump. In addition, a portion of the fuel is bypassed or split after the first stage for the SPLTEX cycle to increase cycle performance. To meet the demands of increased speed, the pumps will use shroudless impellers and hydrostatic bearings.

The risk reduction activities focus on the shroudless impellers, hydrostatic bearing and flow extraction impact on pump performance. The hydrostatic bearings are at TRL=3 and remainder of the pump risk items are at TRL=4. The five-year schedule, Figure 25, and high-speed fuel pump waterfall chart, Figure 29 show the risk reduction plan.

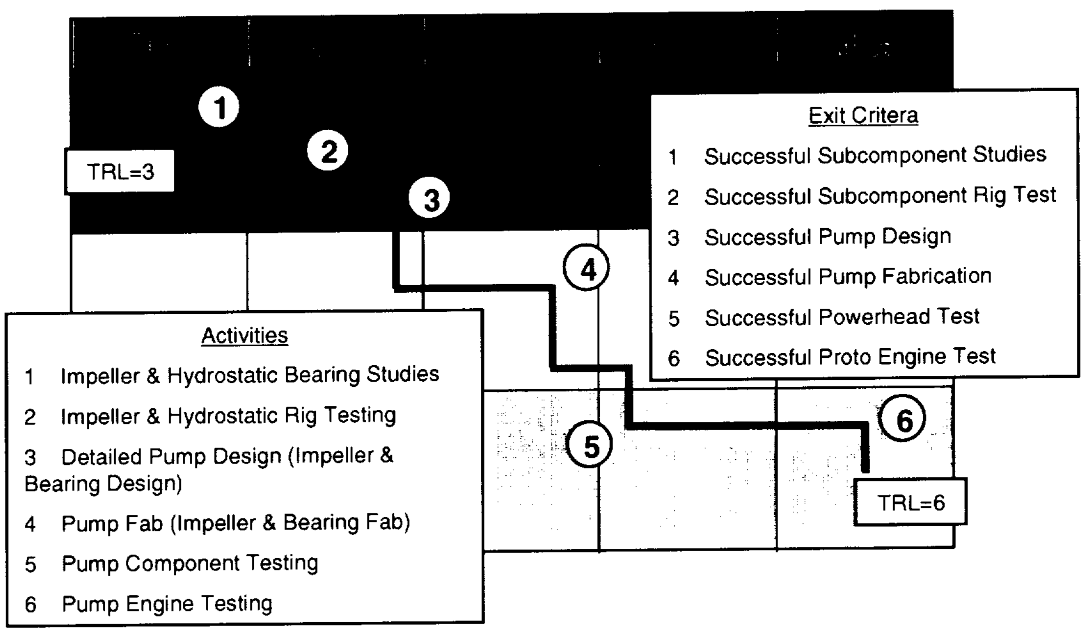


Figure 29 High Speed Fuel Pump Risk Reduction Waterfall Chart

The shroudless impeller and flow extraction risks are reduced through detailed CFD analysis, modeling and component level testing. The hydrostatic bearing risk is mitigated by analyzing bearing dynamic behavior (startup, shutdown and transients) via bearing models anchored in rig testing. These risks will achieve TRL=6 via successful pump test completion.

5.5.1.5 Improved Durability Injectors

Durability improvements for both the main and preburner injectors are based on the same design concept used for the combustion chamber, formed platelets. Careful design of the individual platelets allows an assembly to easily be formed with the necessary passages to distribute the fuel and oxidizer to any desired nozzle configuration. This technique eliminates LOX posts and their attendant reliability issues. Flexibility in injector nozzle configuration allows tailoring to ensure uniform combustion, minimize combustion chamber hot streaks, and optimize transient (startup, shutdown, etc) operation.

Main injector durability benefits are applicable for all three selected cycles. Preburner benefits are applicable to the SBFRSC and SBORSC cycles. The SPLTEX cycle does not require a preburner.

The preburner injector TRL=3 and the main injector TRL=4. Injector risk reduction activities focus on thermal compatibility, combustion stability and producibility as seen in the five-year plan, Figure 25, and the injector risk reduction waterfall chart, Figure 30

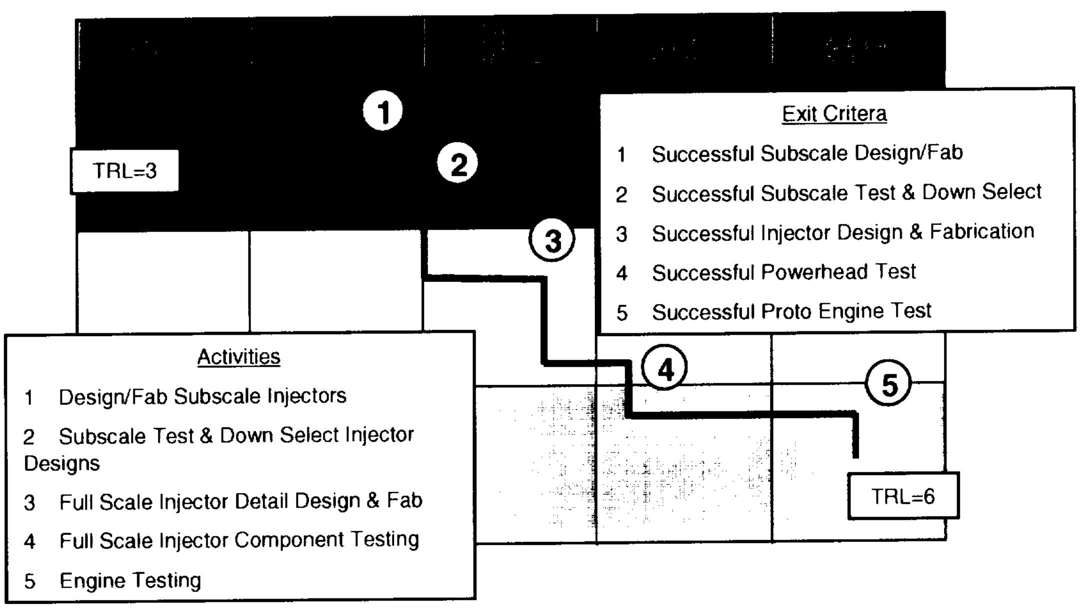


Figure 30 Improved Durability Injector Risk Reduction Waterfall Chart

First step is to conduct subscale combustion testing of several designs of both the preburner and main injectors allowing rapid characterization and optimization of thermal compatibility and transient operation issues. The most successful designs are then carried forward to full scale design and development. Successful testing of the full scale designs advances the technologies to TRL=6.

5.5.1.6 LOX Cooled Nozzle Section

A LOX cooled section was added to the nozzle to replace the LOX heat exchanger used in the baseline SSME engines. The GOX output is used to pressurize the LOX tank and to power the LOX boost pump. Using GOX in place LOX to power the LOX boost pump increases cycle performance by reducing main LOX pump mass flow requirement. This feature is applicable to the SBFRSC and SPLTEX cycles. Other means are used to pressurize the LOX tank in the SBORSC cycle.

Risk reduction activities for this component focuses on fabrication, the consequences of GOX leakage into the nozzle and complications associated with the additional plumbing at the nozzle. The risk reduction plan is shown in the five-year plan, Figure 25 and the LOX cooled nozzle risk reduction waterfall chart, Figure 31.

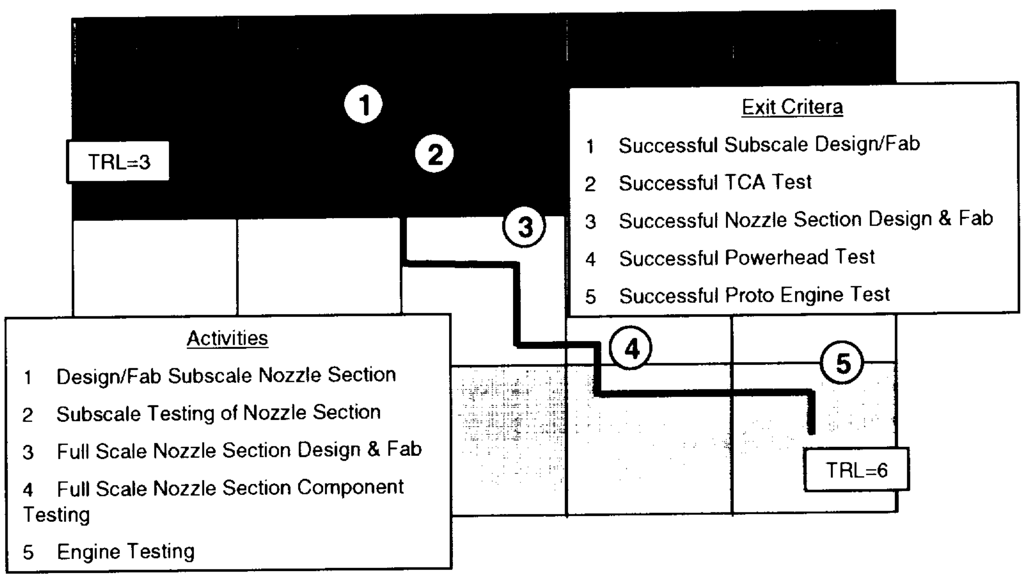


Figure 31 LOX Cooled Nozzle Section Risk Reduction Waterfall Chart

Subscale testing provides a means to confirm fabrication techniques as well as investigation of the safety issues relative to leaks into the nozzle are addressed in subscale testing in the TCA test. Lessons learned from the subscale testing insure successful design and testing of full scale hardware. Successful completion of full scale testing matures the technology to TRL=6.

5.5.1.7 Electro Mechanical / Electro-pneumatic Actuators

Electromechanical and/or electro-pneumatic actuators offer a reliability improvement to the engine by replacing the current hydraulic system with a simpler robust system with built in redundancy. The risk reduction plan for the actuators is shown in the five year plan, Figure 25, and the actuator risk reduction waterfall chart, Figure 32.

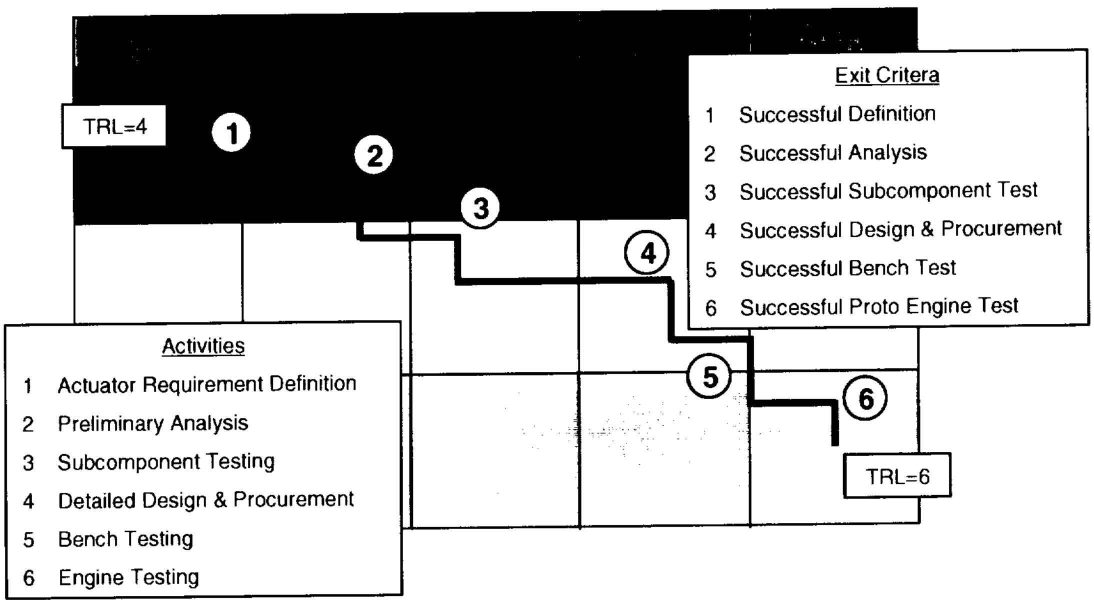


Figure 32 EMA/EPA Actuator Risk Reduction Waterfall Chart

Risk reduction begins with actuator requirement definition. Analysis and subcomponent testing follow. Detail design and procurement are the next step. Once the hardware is completed it will be submitted to bench testing and finally engine level testing at the prototype engine test. Successful completion of the prototype testing will result in a TRL=6.

5.5.1.8 Controller w/ Integrated EHMS

The controller with integrated engine health management system offers substantial improvement to engine reliability. The improvements include:

- Addition of in-flight fault detection and accommodation
- Incorporation of diagnostics, prognostics, and health monitoring functions in the EHMS
- State-of-the-art sensors, including optical pyrometers, plume spectroscopy, acoustic bearing sensors, and infrared engine bay sensors
- Incorporation of Self-Tuning On-board Real-time Modeling (STORM)

Risks are focused in the successful design and implementation of the system into the rocket engine system. The risk reduction plan is shown in the five year plan, Figure 33.

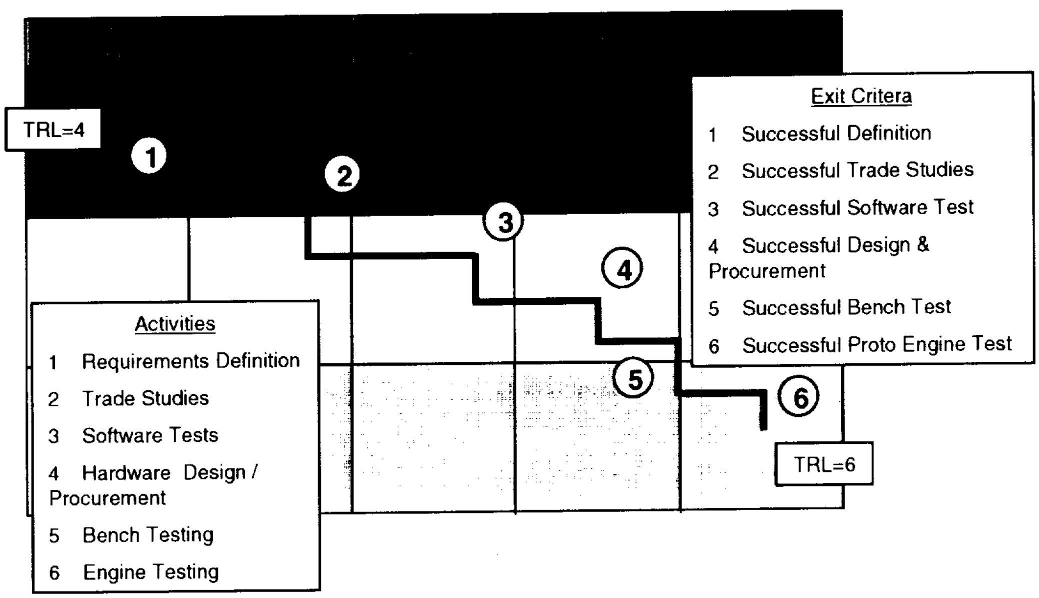


Figure 33 Integrated Controls EHMS Risk Reduction Waterfall Chart

Risk reduction will commence with definition of the controller and health management requirements. Followed by sensor, algorithm and architecture trade studies. Software testing follows the trade studies. Successful completion of software testing will lead to hardware design and procurement. The hardware will then be submitted to bench testing and then engine testing on the prototype engine. Resulting in a TRL= 6 following successful completion of the prototype engine test.

5.5.1.9 External Hot Gas & Material Containment Systems

The intent of the containment systems are to minimize the risk of loss of engine cascading to loss of vehicle. Upon the rare case when an engine fails and either engine hardware or hot gas is released the containment systems will prevent damage to other engines or the vehicle. Since there are no current examples of rocket engine containment systems the first portion of the task will be to survey existing systems such as those used in aircraft and determine how they may be modified to meet the unique space vehicle requirements. Also there are the dual requirements of containing not only material fratricide but also hot gases. Gas leak detection system improvements will be evaluated in the survey and fundamental properties activities. The risk reduction plan is shown in the five-year plan, Figure 25 and the containment system risk reduction waterfall chart, Figure 34.

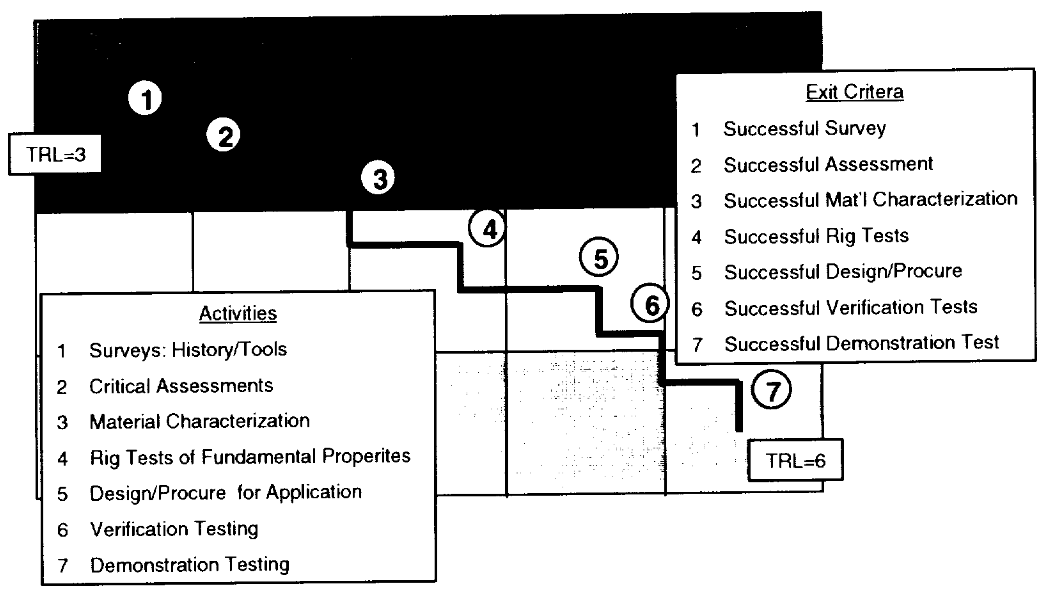


Figure 34 Containment System Risk Reduction Waterfall Chart

As can be seen in the waterfall the first portion of the risk reduction consists of surveys of all technologies that could be useful in creation of a containment system. Then a critical assessment of applicability of the survey results. Following is a step for material characterization to determine hot gas effects (melt, vaporization, etc) and material effects (impact properties under space conditions, etc) for both materials normally used in rocket engines and vehicle systems and also other materials that maybe be better suited to minimize damage. Some expansion of material properties may be required to aid in determining the most appropriate materials. Next are rigs designed to test the most promising materials fundamental properties in expected configurations and environments. Successful completion of these steps leads to design and procurement of the engine containment systems. Thorough verification testing will be conducted culminating in some form of engine demonstration (either real or mockup) which will bring the systems to TRL=6.

5.5.1.10 Hydraulic Fuel Boost Pump Turbine

The SPLTEX cycle includes a fuel boost pump with a hydraulic turbine. While there is no reliability gain, it's a performance enhancement tied to the split flow aspect of the expander design. The risk associated with this turbine is moderate and is focused in successful design and development as shown in the five year plan Figure 25 and the risk reduction waterfall chart, Figure 35.

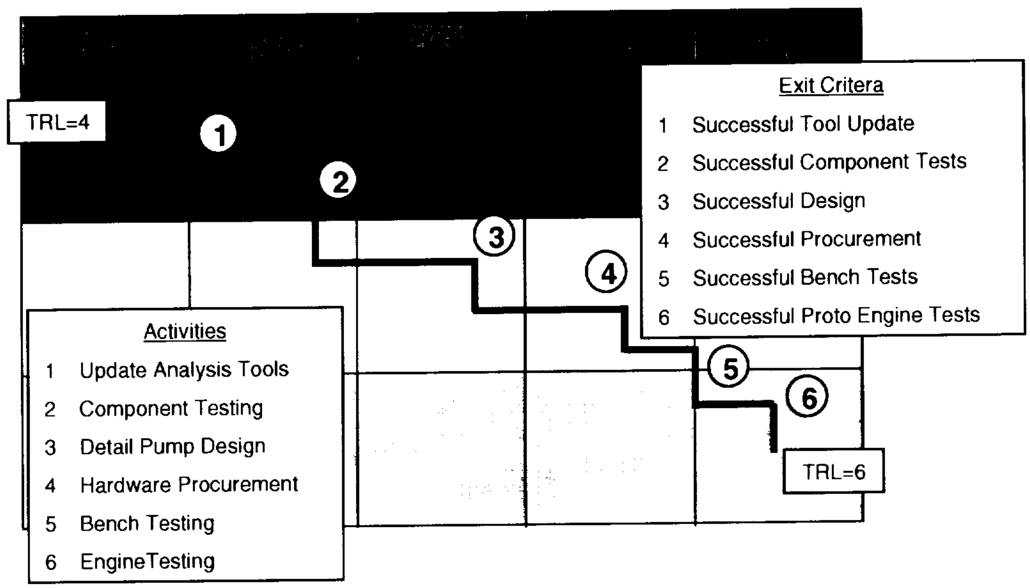


Figure 35 Hydraulic Fuel Boost Pump Turbine Risk Reduction Waterfall Chart

Analytical tools must be enhanced to handle liquid rather than gas as the fluid medium. Some rig testing will be required to validate the modeling tools. Detail design including CFD will then commence followed by procurement. Full scale boost pump testing will be conducted prior to integration into the engine at the prototype engine test. Successful prototype testing will mature the technology to TRL=6.

5.5.1.11 *LOX Boost Pump*

There are some risks associated with unique cycle enhancing performance features found in the LOX boost pumps used for all three of the selected cycles. These features are low supply pressure and GOX/LOX mixing at the pump outlet. Since these are closed cycles the GOX from the turbine outlet is mixed into the LOX from the pump outlet. See the cycle schematics in section 5.1 for details. There is some concern about this mixing but it has been successfully demonstrated in Russian engines. The risk reduction plans are shown in the five year plan, Figure 25 and the LOX boost pump risk reduction waterfall chart, Figure 36.

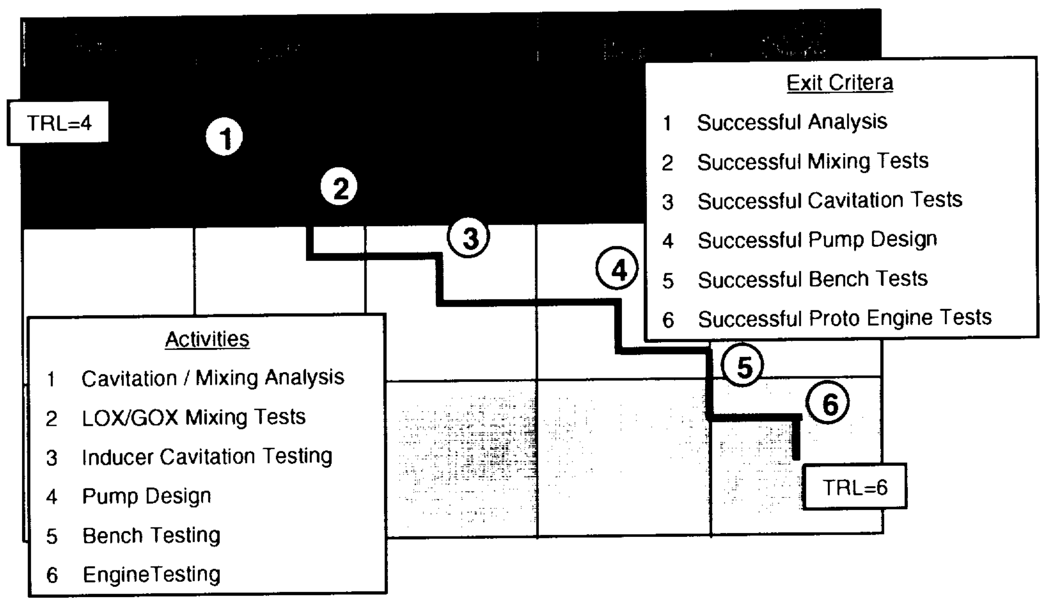


Figure 36 LOX Boost Pump Risk Reduction Waterfall Chart

Low inlet pressure risks are addressed by investigating pump inducer cavitation and expanding the inducer design database to incorporate this knowledge. The inducer design database will also be expanded to incorporate mixed flow (both LOX & GOX). Rig testing will follow to validate the modeling tools. Once this is successfully concluded the detailed pump design will be initiated followed by hardware procurement. The pump will be subjected to bench tests prior to installation on an engine. These technologies will mature to TRL=6 with successful LOX boost pump testing in the prototype engine test program

6.0 SUMMARY

This NRA 8-27 study determined the inherently safest engine cycles, identified and evaluated technologies that enhance these cycles, and proposed a risk reduction methodology and program plan to incorporate these improvements in support of 2nd Gen. RLV architecture and program goals.

The study started by identifying six cycles that showed the potential for high safety and reliability, and that met the needs of the various 2nd Gen. RLV architectures. These cycles, which are described in detail in section 3.1.1, are:

- Dual Burner-Fuel Rich-Staged Combustion (DBFRSC)
- Dual Burner-Full Flow-Staged Combustion (DBFFSC)
- Single Burner-Fuel Rich-Staged Combustion (SBFRSC)
- Single Burner-Fuel Rich-Gas Generator (SBFRGG)
- Split Expander (SPLTEX)
- Single Burner-Oxidizer Rich-Staged Combustion (SBORSC)

A safety analysis was performed on each of these cycles based on the use of currently available state-of-the-art technologies, and a consistent Technology Readiness Level (TRL) of 7. This safety and reliability analysis is discussed in more detail in section 3.2, and a summary chart of Loss of Vehicle (LOV) rates is shown in Figure 37.

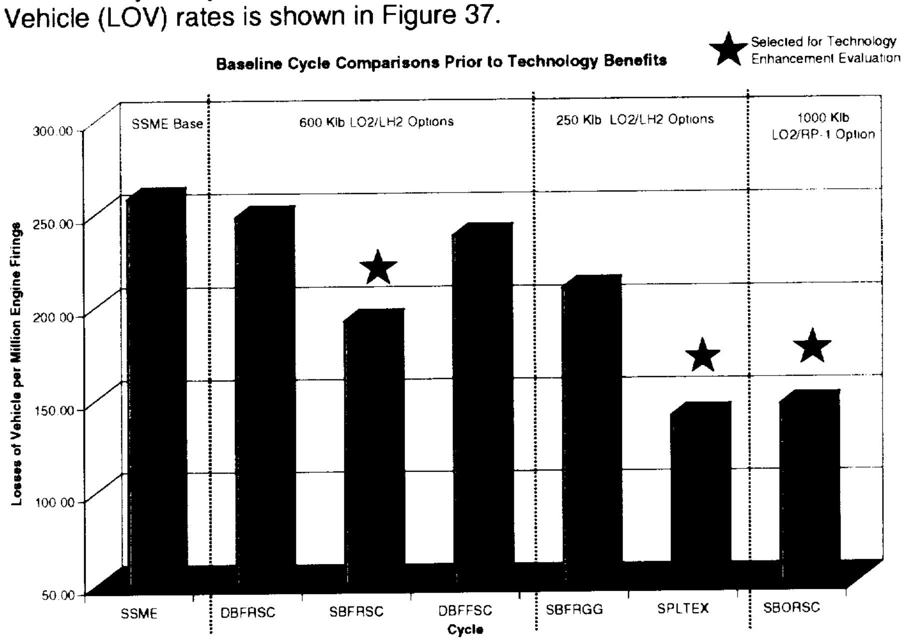


Figure 37 Engine Cycle Impact on LOV Rates

In addition to the safety analysis, cost and performance trade studies were performed and are documented in this report. Selection criteria were established, based upon program goals and objectives. Since flight safety is a primary objective of NASA, it is the primary selection criterion for this analysis.

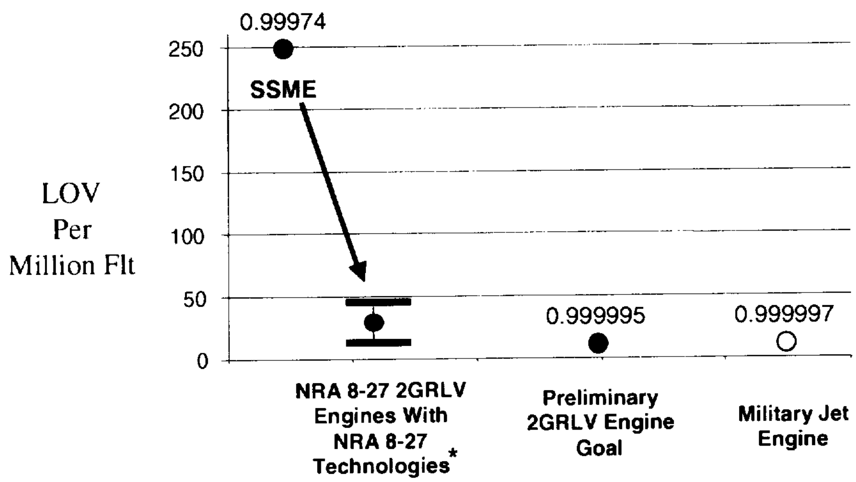
Three cycles, one from each thrust class, were identified as the inherently safest cycles, and selected for further study and analysis. These cycles are:

- Single Burner-Fuel Rich-Staged Combustion (SBFRSC)
- Split Expander (SPLTEX)
- Single Burner-Oxidizer Rich-Staged Combustion (SBORSC)

Next, technology improvements were identified and their effect on key parameters was studied (e.g. their benefit to LOV, production cost, etc). The various technology improvements, their applicability to the selected cycles, and their benefit to key parameters is discussed in sections 4 and 5. Key technologies include:

- Controller with Integrated EHMS
- Improved Durability Combustion Chamber
- Fail Safe Hot Gas System
- Material Containment System
- Gas Containment System
- Milled Channel Nozzle
- High Speed Main Fuel Pump

The selected cycles were evaluated with these technologies, and the resultant safety analysis showed significant improvement over the baseline (SSME). Based on the space Shuttle Quantitative Risk Assessment System (QRAS) data, the loss of vehicle (LOV) rate due to main engine is currently 258 per million missions. Depending upon the assumptions of effectiveness and implementation success of the technologies, the resultant safety of the engines ranges from 5 to 45 LOV events per million firings as shown in the following figure. This analysis is discussed in detail in section 5.2.



LOV Improvement Through Technology Improvements

A key piece of information that needs to be established is the "LOV per million" requirement for each booster engine. This requirement needs to support the NASA goal of system safety of 1 in 1000 LOV. It is our intention to work with the vehicle manufacturers to establish a reasonable requirement based upon their architecture, recognizing that the portion allocated to the propulsion system, as well as the number of booster engines, will change this value.

For the purpose of this study, we have established a preliminary goal of 5 events per million for a booster engine, which we believe is optimistic but achievable. As mentioned above, the level achievable by incorporating the current technology list is between 5 and 45 events per million, based upon the assumptions of effectiveness and implementation success of the various technologies. Contained within this report are the program plans to incorporate the various technologies as well as the overall system program plan. It is important to note that as the goal becomes more aggressive, the cost of the program increases, since additional technologies will need to be pursued to accommodate meeting a more aggressive goal.

Some additional technologies that have been identified include:

- Engine Hardware
  Long life coatings / Plating
  Advanced materials (discontinuously reinforced aluminum, etc)
  Additional advanced controller/EHMS sensors

- Engine Operations:
  Thorough engine diagnostics via controller/EHMS during short duration pre-launch hold down.
  Engine throttling when applicable during boost phase to minimize failures
  Optimize engine shutdown to improve reliability

- Vehicle / Engine Interactivity
  Better communication between engine and vehicle to allow optimized engine operation to minimize failures

In summary, this study has identified the three safest booster engine cycles, identified the technologies and program plans to incorporate these technology improvements, and shown that the resultant safety output should be in the correct range to support the architecture safety requirements.

7.0 APPENDIX A – BASELINE CYCLE OPERATING CONDITIONS

A.1. Baseline DBFRSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	600000.
CHAMBER PRESSURE (PSIA)	3000.0
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	6.036
ENG. FLOW RATE (LBM/S)	1334.05
DEL. VACUUM ISP (SEC)	450.90
DEL. SL ISP (SEC)	380.35
CORE AREA RATIO	59.51
SL Thrust (LBF)	506126.

** MAIN PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	FUEL PUMP 3RD STAGE	LOX PUMP MAIN	LOX PUMP PREBURNER
INLET TOT PRESS(PSIA)	275.7	2196.2	4207.3	389.4	4078.4
INLET TOTAL TEMP (R)	41.8	61.1	79.8	167.9	184.8
MASS FLOW (LBM/S)	190.98	190.98	190.98	1350.59	113.84
EFFICIENCY	0.7856	0.7803	0.7777	0.7955	0.7404
SPEED (RPM)	33263.	33263.	33263.	29164.	29164.
TIP SPEED (FT/S)	1928.	1928.	1928.	793.	648.

** BOOST PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	30.0	100.0
INLET TOTAL TEMP (R)	37.0	164.0
MASS FLOW (LBM/S)	190.58	1143.47
EFFICIENCY	0.7592	0.8081
SPEED (RPM)	19702.	5647.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	5169.6	5169.6	4035.9	4435.9
FLNG IN TOT TEMP (R)	1733.0	1152.1	184.9	411.5
MASS FLOW (LBM/S)	176.58	69.60	187.27	39.13
FLANGE TO FLANGE EFF.	0.8286	0.8154	0.7029	0.6933
PR RATIO (FLANGE-T/T)	1.570	1.570	7.899	1.261
SPEED (RPM)	33263.	29166.	5647.	19702.

** ENGINE PHYSICAL CHARACTERISTICS **

TOTAL ENGINE LENGTH (IN)	164.3
NOZZLE EXIT DIAMETER (IN)	90.1
TOTAL ENGINE WEIGHT (LBM)	8569.5
THRUST-TO-WEIGHT	70.0

A.2. Baseline DBFFSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	600000.
CHAMBER PRESSURE (PSIA)	3000.0
SEA LEVEL THRUST (LBF)	506359.
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	6.023
ENG. FLOW RATE (LBM/S)	1330.79
DEL. VACUUM ISP (SEC)	451.12
SEA LEVEL ISP (SEC)	380.71
TOTAL AREA RATIO	59.42

** MAIN PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	FUEL PUMP 3RD STAGE	LOX PUMP MAIN
INLET TOT PRESS(PSIA)	187.0	2278.7	4524.1	389.4
INLET TOTAL TEMP (R)	49.6	74.8	97.9	167.9
MASS FLOW (LBM/S)	222.59	222.59	190.11	1266.60
EFFICIENCY	0.7919	0.7867	0.7705	0.7936
SPEED (RPM)	36000.	36000.	36000.	34345.
TIP SPEED (FT/S)	2119.	2119.	2119.	928.

** BOOST PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	30.0	100.0
INLET TOTAL TEMP (R)	37.0	164.0
MASS FLOW (LBM/S)	190.11	1140.68
EFFICIENCY	0.7777	0.8113
SPEED (RPM)	16155.	5167.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	4816.4	4795.0	5810.1	4297.9
FLNG IN TOT TEMP (R)	1405.8	1250.0	193.0	99.9
MASS FLOW (LBM/S)	285.61	1038.52	126.17	32.48
FLANGE TO FLANGE EFF.	0.8429	0.8197	0.6900	0.3050
PR RATIO (FLANGE-T/T)	1.478	1.427	11.371	16.832
SPEED (RPM)	36000.	34345.	5167.	0.

** ENGINE PHYSICAL CHARACTERISTICS **

TOTAL ENGINE LENGTH	(IN)	164.2
NOZZLE EXIT DIAMETER	(IN)	90.1
TOTAL ENGINE WEIGHT	(LBM)	8234.7
THRUST-TO-WEIGHT		72.9

A.3. Baseline SBFRSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	600000.
SL THRUST (LBF)	506292.
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	6.035
CHAMBER PRESSURE (PSIA)	3000.0
ENG. FLOW RATE (LBM/S)	1331.29
DEL. VACUUM ISP (SEC)	451.65
DEL. SL ISP (SEC)	381.11
THROAT AREA (IN2)	107.20
TOTAL AREA RATIO	59.50
DESIGN AREA RATIO	87.94

** MAIN PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	FUEL PUMP 3RD STAGE	LOX PUMP MAIN	LOX PUMP PREBURNER
INLET TOT PRESS(PSIA)	275.7	2181.1	4192.8	389.4	4078.4
INLET TOTAL TEMP (R)	40.5	58.0	74.8	167.9	184.8
MASS FLOW (LBM/S)	190.18	190.18	190.18	1349.85	107.25
EFFICIENCY	0.8086	0.8063	0.8048	0.7955	0.7150
POWER (HP)	25577.5	25947.2	26242.6	23912.3	2245.0
SPEED (RPM)	43927.	43927.	43927.	29172.	29172.
TIP SPEED (FT/S)	2000.	2000.	2000.	793.	692.

** BOOST PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	30.0	100.0
INLET TOTAL TEMP (R)	37.2	164.0
MASS FLOW (LBM/S)	190.18	1141.11
EFFICIENCY	0.7880	0.8081
POWER (HP)	4111.0	1729.1
SPEED (RPM)	16218.	5652.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	5542.7	5542.7	4035.9	4271.7
FLNG IN TOT TEMP (R)	1313.3	1313.3	184.9	941.5
MASS FLOW (LBM/S)	177.07	62.42	186.9	40.37
FLANGE TO FLANGE EFF.	0.8295	0.7913	0.6980	0.5633
PR RATIO (FLANGE-T/T)	1.68	1.68	7.90	1.14
POWER (HP)	77767.3	26157.3	1729.1	4111.0
SPEED (RPM)	43927.	29172.	5652.	16218.

** ENGINE PHYSICAL CHARACTERISTICS **

CHAMBER LENGTH	(IN)	29.0
TOTAL NOZZLE LENGTH	(IN)	123.3
TOTAL ENGINE LENGTH	(IN)	164.3
NOZZLE EXIT DIAMETER	(IN)	90.1
TOTAL ENGINE WEIGHT	(LBM)	7950.4
THRUST-TO-WEIGHT		75.5

A.4. Baseline SBFRGG Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	250000.
SEA LEVEL THRUST	201564.
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	7.682
CHAMBER PRESSURE (PSIA)	2250.0
ENG. FLOW RATE (LBM/S)	612.67
DEL. VACUUM ISP (SEC)	408.91
SEA LEVEL ISP (SEC)	329.68
THROAT AREA (IN2)	57.57
TOTAL AREA RATIO	54.85
DESIGN AREA RATIO	81.21

** PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	FUEL PUMP 3RD STAGE	LOX PUMP
INLET TOT PRESS(PSIA)	30.0	1329.6	2660.7	100.0
INLET TOTAL TEMP (R)	37.0	51.0	65.1	164.0
MASS FLOW (LBM/S)	87.52	87.52	87.52	524.90
EFFICIENCY	0.7568	0.7485	0.7232	0.7879
POWER (HP)	8698.3	8824.6	9154.8	7504.7
SPEED (RPM)	33812.	33812.	33812.	17132.
TIP SPEED (FT/S)	1570.	1570.	1570.	632.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB
FLNG IN TOT PRS(PSIA)	2123.2	2123.2
FLNG IN TOT TEMP (R)	1900.3	1900.3
MASS FLOW (LBM/S)	24.05	15.96
FLANGE TO FLANGE EFF.	0.5602	0.3558
PR RATIO (FLANGE-T/T)	7.09	3.28
POWER (HP)	26677.7	7504.7
SPEED (RPM)	33812.	17132.

** ENGINE PHYSICAL CHARACTERISTICS **

CHAMBER LENGTH	(IN)	29.0
TOTAL NOZZLE LENGTH	(IN)	79.8
TOTAL ENGINE LENGTH	(IN)	120.8
NOZZLE EXIT DIAMETER	(IN)	63.4
TOTAL ENGINE WEIGHT	(LBM)	3895.4
THRUST-TO-WEIGHT		64.2

A.5. Baseline SPLTEX Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	250636.
SEA LEVEL THRUST (LBF)	202280.
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	6.288
CHAMBER PRESSURE (PSIA)	1500.0
ENG. FLOW RATE (LBM/S)	575.00
DEL. VACUUM ISP (SEC)	435.89
SEA LEVEL ISP (SEC)	351.79
THROAT AREA (IN2)	88.93
TURB BYPASS RATIO (%)	5.0
CORE AREA RATIO	36.25
TOTAL AREA RATIO	36.25
DESIGN AREA RATIO	54.53

** PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	FUEL PUMP 3RD STAGE	LOX PUMP	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	171.4	1880.9	3823.4	234.7	30.0	100.0
INLET TOTAL TEMP (R)	40.7	57.9	79.6	166.2	37.4	162.9
MASS FLOW (LBM/S)	82.14	41.70	41.70	616.62	82.14	492.86
EFFICIENCY	0.7886	0.7214	0.7187	0.7992	0.5773	0.7158
POWER (HP)	10352.1	6336.1	6368.7	5354.1	1431.9	407.0
SPEED (RPM)	50000.	50000.	50000.	19066.	21981.	3618.
TIP SPEED (FT/S)	1844.	1844.	1844.	556.	786.	169.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	4306.3	2044.0	2113.6	1629.5
FLNG IN TOT TEMP (R)	800.0	702.9	174.1	297.0
MASS FLOW (LBM/S)	39.61	39.50	123.97	3.68
FLANGE TO FLANGE EFF.	0.7104	0.7768	0.5000	0.4624
PR RATIO (FLANGE-T/T)	2.10	1.19	6.86	16.30
POWER (HP)	23056.9	5354.1	407.0	1431.9
SPEED (RPM)	50000.	19066.	3618.	21981.

** ENGINE PHYSICAL CHARACTERISTICS **

CHAMBER LENGTH	(IN)	33.0
CORE ENGINE LENGTH	(IN)	125.0
TOTAL NOZZLE LENGTH	(IN)	80.0
TOTAL ENGINE LENGTH	(IN)	125.0
NOZZLE EXIT DIAMETER	(IN)	64.1
TOTAL ENGINE WEIGHT	(LBM)	3467.7
THRUST-TO-WEIGHT		72.3

A.6. Baseline SBORSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	999999.
SEA LEVEL THRUST	830784.
INLET MIXTURE RATIO	2.720
CHAMBER MIXTURE RATIO	2.719
CHAMBER PRESSURE (PSIA)	3500.0
ENG. FLOW RATE (LBM/S)	2824.40
DEL. VACUUM ISP (SEC)	354.06
DEL. SEA LVL ISP (SEC)	294.15
THROAT AREA (IN2)	152.56
TOTAL AREA RATIO	75.47
DESIGN AREA RATIO	111.00

** PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP KICK STG.	LOX PUMP	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	183.9	6891.4	339.4	70.0	100.0
INLET TOTAL TEMP (R)	530.5	574.5	186.5	529.9	164.0
MASS FLOW (LBM/S)	801.66	43.13	2118.87	759.25	2065.15
EFFICIENCY	0.7330	0.3627	0.7990	0.7120	0.5377
POWER (HP)	45447.4	1673.2	73348.7	659.1	4219.4
SPEED (RPM)	18500.	18500.	18500.	2951.	3525.
TIP SPEED (FT/S)	1135.	640.	1012.	0.	269.

** TURBINE PERFORMANCE CHARACTERISTICS **

	MAIN TURB	LOX TURB BOOST	FUEL TUR BOOST
FLNG IN TOT PRS(PSIA)	6776.2	3745.5	8302.9
FLNG IN TOT TEMP (R)	1550.0	1395.3	567.6
MASS FLOW (LBM/S)	2104.79	53.72	42.42
FLANGE TO FLANGE EFF.	0.7709	0.3695	0.0000
PR RATIO (FLANGE-T/T)	1.79	9.42	39.57
POWER (HP)	120469.4	4219.4	659.1
SPEED (RPM)	18500.	3525.	2951.

** ENGINE PHYSICAL CHARACTERISTICS **

CHAMBER LENGTH	(IN)	14.0
NOZZLE LENGTH	(IN)	154.9
ENGINE LENGTH	(IN)	180.9
NOZZLE EXIT DIAMETER	(IN)	121.1
TOTAL ENGINE WEIGHT	(LBM)	13533.0
THRUST-TO-WEIGHT		74.0

8.0 APPENDIX B -- TECHNOLOGY IMPROVEMENTS IMPACTS ON CYCLE PERFORMANCE

SBFRSC Technology Improvements Cycle Performance Impacts

Parameter List	**Basepoint**	1 LOX-cooled nozzle	2 Increased Component Efficiencies	[redacted]	4 High speed fuel pump	5 Low LOX inlet pressure	[redacted]
Vacuum Specific Impulse (sec)	**451.7**	451 7	451.7		451 7	451 7	
Sea Level Specific Impulse (sec)	**381.1**	381 1	381 1		381 3	381 1	
Turbine Temp (deg. R)	**1334**	1287	1257		1354	1348	
Fuel Pump Speed (rpm)	**36557**	36179	35557		45070	35557	
Fuel Pump Tip Speed (ft/s)	**1936**	1940	1931		2346	1936	
Fuel Pump AN^2 (in^2*rpm^2*10^(-8))	**400**	400	400		570	400	
No. of fuel pump stages	**3**	3	3		2	3	
Delta-P for chamber coolant (psid)	**1247**	1247	1247		1247	1247	
Pexit for fuel pump (psia)	**6300**	6300	6300		6300	6300	
Pexit for LOX pump (psia)	**8214**	8214	8214		8213	8214	
Exit temp of chamber coolant (deg. R)	**385**	398	384		386	385	
Fuel Pump DN (mm*RPM*10^(-6))	**3.4**	3 5	3 4		4 0	3 4	
Bearing Type	**Conv.**	Conv.	Conv.		Conv.	Conv.	
Single- or dual-position nozzle	**Single**	Single	Single		Single	Single	
Chamber construction (milled channel, FPL or tube	**Tubes**	Tubes	Tubes		Tubes	Tubes	
Hydraulic or gas fuel boost turbine	**Gas**	Gas	Gas		Gas	Gas	
Nozzle construction (milled channel, FPL or tubes)	**Tubes**	Tubes	Tubes		Tubes	Tubes	

Parameter List	[redacted]	8 Increased combustion efficiency	9 FPL chamber	10 Milled Channel Nozzle
Vacuum Specific Impulse (sec)		452 6	451 7	451 7
Sea Level Specific Impulse (sec)		381 7	381.1	381 1
Turbine Temp (deg. R)		1334	1441	1297
Fuel Pump Speed (rpm)		35557	35557	35557
Fuel Pump Tip Speed (ft/s)		1936	1936	1936
Fuel Pump AN^2 (in^2*rpm^2*10^(-8))		400	400	400
No. of fuel pump stages		3	3	3
Delta-P for chamber coolant (psid)		1247	1018	1247
Pexit for fuel pump (psia)		6300	6300	6300
Pexit for LOX pump (psia)		8214	8214	8214
Exit temp of chamber coolant (deg. R)		385	488	413
Fuel Pump DN (mm*RPM*10^(-6))		3.4	3 4	3 4
Bearing Type		Conv.	Conv.	Conv
Single- or dual-position nozzle		Single	Single	Single
Chamber construction (milled channel, FPL or tube		Tubes	FPL	Tubes
Hydraulic or gas fuel boost turbine		Gas	Gas	Gas
Nozzle construction (milled channel, FPL or tubes)		Tubes	Tubes	Milled

SPLTEX Technology Improvements Cycle Performance Impacts

Parameter List	**Base**	1 LOX-cooled nozzle	2 Increased Component Efficiencies	3 Hydraulic fuel boost turbine	4 High speed fuel pump	5 Low LOX inlet pressure	6 Copper tubular chamber
Turbine Temp (deg. R)	**800**	800	800	800	800	800	800
Fuel Pump Speed (rpm)	**50000**	50000	50000	**100000**	**110000**	50000	50000
Fuel Pump Tip Speed (ft/s)	**1844**	1859	**1754**	**2428**	**2669**	1888	**1754**
Fuel Pump AN^2 (in^2*rpm^2*10^(-8)	**237**	225	251	**593**	**676**	231	277
No. of fuel pump stages	**3**	3	3	**2**	**2**	3	3
Delta-P for chamber coolant (psid)	**734**	734	734	734	734	734	734
Pexit for fuel pump (psia)	**5810**	**6012**	**5200**	**6057**	**7327**	**6142**	**5117**
Pexit for LOX pump (psia)	**2124**	2140	2124	2124	2124	2124	2124
Exit temp of chamber coolant (deg. R)	**800**	800	800	800	800	800	800
Fuel Pump DN	**2.6 M**	2.6 M	2.6 M	**NA**	**NA**	2.6 M	2.7 M
Bearing Type	**Roller**	Roller	Roller	**Hydrostatic**	**Hydro St.**	Roller	Roller
Single- or dual-position nozzle	**S**	S	S	S	S	S	S
Chamber construction (milled channel, FPL or tubes)	**M/C**	M/C	M/C	M/C	M/C	M/C	**Tube**
Hydraulic or gas fuel boost turbine	**Gas**	Gas	Gas	**Hyd.**	Gas	Gas	Gas
Sea Level Isp (sec)	**351.8**	351.8	351.8	354.6	351.8	351.8	351.8
Vacuum Isp (sec)	**435.9**	435.9	435.9	433.7	435.9	435.9	435.9
Nozzle Construction (Milled-channel, FPL, or tubes)	**Tube**	Tube	Tube	Tube	Tube	Tube	Tube

Parameter List	7 Split Circuit Cooling	8 Increased combustion efficiency	9 FPL chamber	10 Milled Channel Nozzle
Turbine Temp (deg. R)	800	800	800	800
Fuel Pump Speed (rpm)	50000	50000	50000	50000
Fuel Pump Tip Speed (ft/s)	**1753**	1844	**1753**	**1964**
Fuel Pump AN^2 (in^2*rpm^2*10^(-8)	246	237	301	223
No. of fuel pump stages	3	3	3	3
Delta-P for chamber coolant (psid)	484	734	734	734
Pexit for fuel pump (psia)	**5144**	5810	**5063**	**6740**
Pexit for LOX pump (psia)	2124	2124	2124	2124
Exit temp of chamber coolant (deg. R)	800	800	800	800
Fuel Pump DN	2.6 M	2.6 M	2.8 M	2.6 M
Bearing Type	Roller	Roller	Roller	Roller
Single- or dual-position nozzle	S	S	S	S
Chamber construction (milled channel, FPL or tubes)	M/C	M/C	**FPL**	M/C
Hydraulic or gas fuel boost turbine	Gas	Gas	G	Gas
Sea Level Isp (sec)	351.8	351.8	351.8	351.8
Vacuum Isp (sec)	435.9	435.9	435.9	435.9
Nozzle Construction (Milled-channel, FPL, or tubes)	Tube	Tube	Tube	M/C

SBORSC Technology Improvements Cycle Performance Impacts

Parameter List	**Basepoint**	[redacted]	2 Increased Component Efficiencies	[redacted]	5 Low LOX inlet pressure	[redacted]
Turbine Temp (deg. R)	**1,550**		1,550		1,550	
Fuel Pump Speed (rpm)	**18,500**		18,000		18,500	
Fuel Pump Tip Speed (ft/s)	**1,135**		1,131		1,135	
Fuel Turbine AN^2 (in^2*rpm^2*10^(-8)	**192**		187		194	
No. of fuel pump stages	**1 + 1 Kick**		1 + 1 Kick		1 + 1 Kick	
Delta-P for chamber coolant (psid)	**1,359**		1,359		1,359	
Pexit for fuel pump (psia)	**8,303**		8,303		8,303	
Pexit for fuel pump kick stage (psia)	**9,619**		8,926		9,649	
Pexit for LOX pump (psia)	**7,713**		7,157		7,737	
Exit temp of chamber coolant (deg. R)	**689**		686		689	
Fuel Pump DN * 10^(-6)	**1.23**		1.19		1.24	
Bearing Type	**Convent'l**		Convent'l		Convent'l	
Single- or dual-position nozzle	**Single**		Single		Single	
Chamber construction (milled channel, FPL or tubes)	**Milled**		Milled		Milled	
Hydraulic or gas fuel boost turbine	**Hydraulic**		Hydraulic		Hydraulic	
Sea Level Specific Impulse	**294.2**		294.2		294.1	
Vacuum Specific Impulse	**354.1**		354.1		354.1	

Parameter List	[redacted]	8 Increased combustion efficiency	[redacted]
Turbine Temp (deg. R)		1,550	
Fuel Pump Speed (rpm)		18,500	
Fuel Pump Tip Speed (ft/s)		1,135	
Fuel Turbine AN^2 (in^2*rpm^2*10^(-8)		191	
No. of fuel pump stages		1 + 1 Kick	
Delta-P for chamber coolant (psid)		1,359	
Pexit for fuel pump (psia)		8,303	
Pexit for fuel pump kick stage (psia)		9,621	
Pexit for LOX pump (psia)		7,714	
Exit temp of chamber coolant (deg. R)		689	
Fuel Pump DN * 10^(-6)		1.23	
Bearing Type		Convent'l	
Single- or dual-position nozzle		Single	
Chamber construction (milled channel, FPL or tubes)		Milled	
Hydraulic or gas fuel boost turbine		Hydraulic	
Sea Level Specific Impulse		296.0	
Vacuum Specific Impulse		355.9	

9.0 APPENDIX C--CYCLE INFORMATION FOR THREE SELECTED CYCLES

C.1. Improved SBFRSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST	(LBF)	600000.
SL THRUST	(LBF)	506464.
INLET MIXTURE RATIO		6.000
CHAMBER MIXTURE RATIO		6.036
CHAMBER PRESSURE	(PSIA)	3000.00
ENG. FLOW RATE	(LBM/S)	1328.66
DEL. VACUUM ISP	(SEC)	452.56
DEL. SL ISP	(SEC)	382.01
THROAT AREA	(IN2)	106.99
TOTAL AREA RATIO		59.51
DESIGN AREA RATIO		87.94

** MAIN PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	LOX PUMP MAIN	LOX PUMP PREBURNER
INLET TOT PRESS(PSIA)	275.7	3203.8	389.4	4078.4
INLET TOTAL TEMP (R)	40.5	67.2	178.1	195.3
MASS FLOW (LBM/S)	189.81	189.81	1186.04	137.13
EFFICIENCY	0.7980	0.8082	0.8173	0.7469
POWER (HP)	39167.5	42937.5	20871.4	2778.1
SPEED (RPM)	87700.	87700.	30892.	30892.
TIP SPEED (FT/S)	2706.	2706.	802.	708.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	5542.7	5542.7	7667.7	4499.8
FLNG IN TOT TEMP (R)	1391.4	1391.4	950.0	941.6
MASS FLOW (LBM/S)	184.53	53.69	28.00	44.37
FLANGE TO FLANGE EFF.	0.8162	0.8053	0.5878	0.5882
PR RATIO (FLANGE-T/T)	1.68	1.68	9.05	1.12
POWER (HP)	82105.0	23649.4	2245.9	4102.3
SPEED (RPM)	87700.	30892.	4889.	16218.

** ENGINE PHYSICAL CHARACTERISTICS **

GIMBAL DISTANCE	(IN)	12.0
CHAMBER LENGTH	(IN)	29.0
TOTAL NOZZLE LENGTH	(IN)	123.1
TOTAL ENGINE LENGTH	(IN)	164.1
NOZZLE EXIT DIAMETER	(IN)	90.0
TOTAL ENGINE WEIGHT	(LBM)	7476.0
THRUST-TO-WEIGHT		80.3

C.2. Improved SPLTEX Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	252888.
SEA LEVEL THRUST (LBF)	205031.
INLET MIXTURE RATIO	6.000
CHAMBER MIXTURE RATIO	6.080
CHAMBER PRESSURE (PSIA)	1500.0
ENG. FLOW RATE (LBM/S)	575.00
DEL. VACUUM ISP (SEC)	439.81
SEA LEVEL ISP (SEC)	356.58
THROAT AREA (IN2)	89.92
TURB BYPASS RATIO (%)	5.0
TOTAL AREA RATIO	35.47
DESIGN AREA RATIO	53.42

** PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP 2ND STAGE	LOX PUMP	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	187.9	2359.4	206.7	30.0	30.0
INLET TOTAL TEMP (R)	41.0	65.2	181.5	37.4	161.9
MASS FLOW (LBM/S)	82.14	58.66	508.50	82.14	492.86
EFFICIENCY	0.7510	0.8150	0.8212	0.5821	0.5633
POWER (HP)	13840.1	10564.2	4522.0	1581.3	641.9
SPEED (RPM)	100000.	100000.	19626.	23677.	5162.
TIP SPEED (FT/S)	2334.	2334.	570.	825.	215.

** TURBINE PERFORMANCE CHARACTERISTICS **

	FUEL TURB	LOX TURB	LOX TURB BOOST	FUEL TURB BOOST
FLNG IN TOT PRS(PSIA)	3742.7	1937.4	1168.7	2347.6
FLNG IN TOT TEMP (R)	800.0	709.6	949.0	65.3
MASS FLOW (LBM/S)	45.88	45.77	15.85	32.85
FLANGE TO FLANGE EFF.	0.7355	0.7699	0.3794	0.4100
PR RATIO (FLANGE-T/T)	1.92	1.13	5.07	1.11
POWER (HP)	24404.3	4522.0	641.9	175.7
SPEED (RPM)	100000.	19626.	5162.	23677.

** ENGINE PHYSICAL CHARACTERISTICS **

CHAMBER LENGTH	(IN)	33.0
TOTAL NOZZLE LENGTH	(IN)	80.0
TOTAL ENGINE LENGTH	(IN)	125.0
NOZZLE EXIT DIAMETER	(IN)	63.7
TOTAL ENGINE WEIGHT	(LBM)	3003.3
THRUST-TO-WEIGHT		84.2

C.3. Improved SBORSC Cycle Output

* ENGINE PERFORMANCE PARAMETERS *

VACUUM THRUST (LBF)	1000000.
SEA LEVEL THRUST	831631.
INLET MIXTURE RATIO	2.720
CHAMBER MIXTURE RATIO	2.718
CHAMBER PRESSURE (PSIA)	3500.0
ENG. FLOW RATE (LBM/S)	2810.06
DEL. VACUUM ISP (SEC)	355.86
DEL. SEA LVL ISP (SEC)	295.95
THROAT AREA (IN2)	151.79
CORE AREA RATIO	75.48
TOTAL AREA RATIO	75.48
DESIGN AREA RATIO	111.00

** PUMP PERFORMANCE CHARACTERISTICS **

	FUEL PUMP 1ST STAGE	FUEL PUMP KICK STG.	LOX PUMP	FUEL PUMP BOOST	LOX PUMP BOOST
INLET TOT PRESS(PSIA)	183.9	6891.4	358.5	70.0	30.0
INLET TOTAL TEMP (R)	530.5	571.5	188.0	529.9	164.3
MASS FLOW (LBM/S)	797.66	43.24	2111.40	755.39	2054.67
EFFICIENCY	0.7630	0.3769	0.8290	0.7120	0.6230
VOL FLOW RATE (GPM)	7111.1	0.0	14148.6	6736.2	13007.9
POWER (HP)	43391.0	1197.6	64968.2	655.7	4752.9
SPEED (RPM)	18000.	18000.	18000.	2959.	3822.
TIP SPEED (FT/S)	1131.	561.	979.	0.	283.

** TURBINE PERFORMANCE CHARACTERISTICS **

	MAIN TURB	LOX TURB BOOST	FUEL TUR BOOST
FLNG IN TOT PRS(PSIA)	6284.3	3745.5	8302.9
FLNG IN TOT TEMP (R)	1550.0	1408.9	564.5
MASS FLOW (LBM/S)	2097.63	56.73	42.27
FLANGE TO FLANGE EFF.	0.8009	0.3975	0.0000
PR RATIO (FLANGE-T/T)	1.66	8.92	39.57
POWER (HP)	109556.8	4752.9	655.7
SPEED (RPM)	18000.	3822.	2959.

** ENGINE PHYSICAL CHARACTERISTICS **

GIMBAL DISTANCE	(IN)	12.0
CHAMBER LENGTH	(IN)	14.0
NOZZLE LENGTH	(IN)	154.5
ENGINE LENGTH	(IN)	180.5
NOZZLE EXIT DIAMETER	(IN)	120.8
TOTAL ENGINE WEIGHT	(LBM)	13533.0
THRUST-TO-WEIGHT		73.9

10.0 APPENDIX D – BASELINE SSME CYCLE RELIABILITY

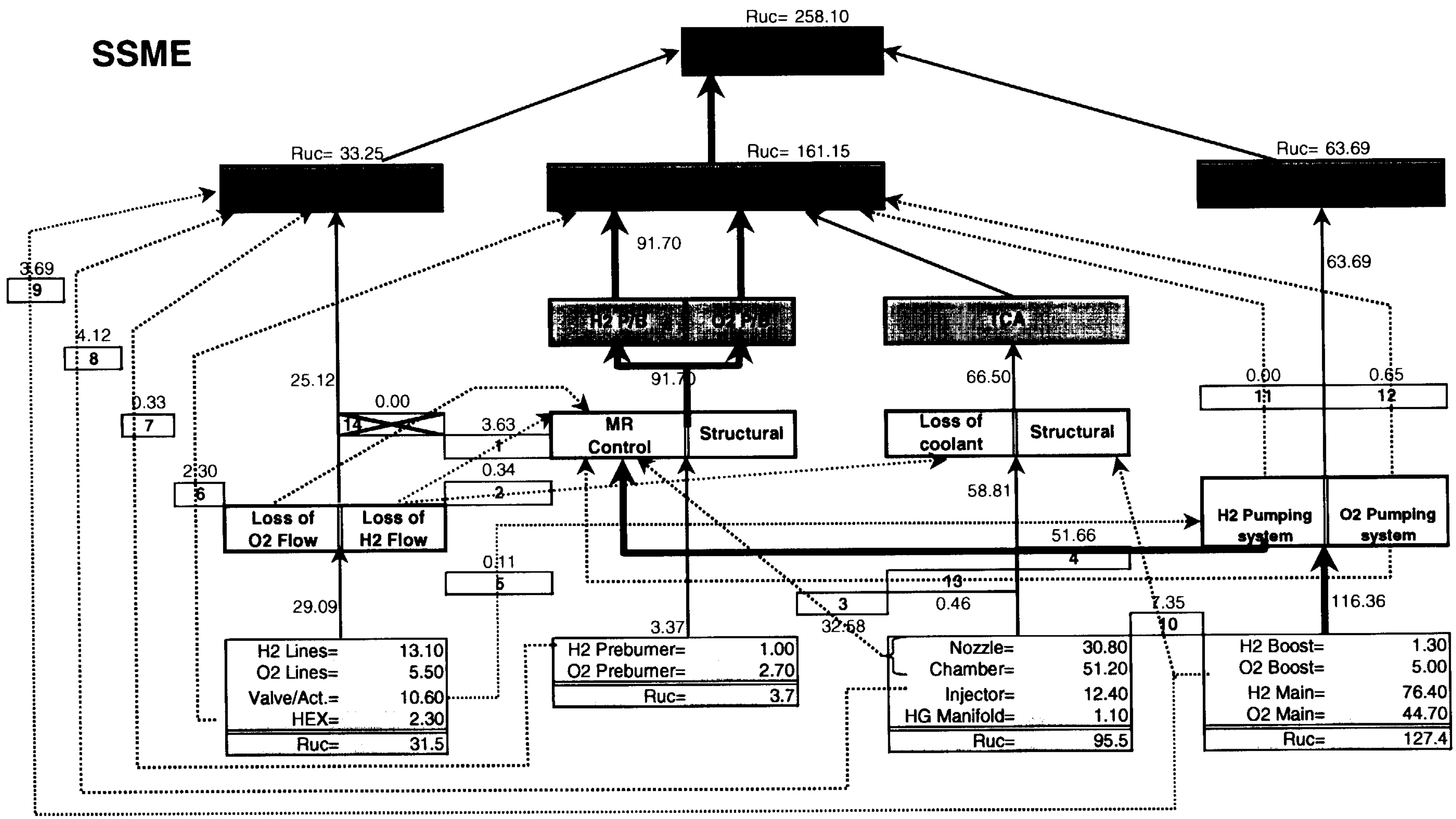
	BASELINE SSME			
	(Rf) Failure Rate	(Rm) Maintenance Rate	(Rs) Shutdown Rate	(Ruc) Uncontained Rate
Mission Reliability (80% Confidence) **	0.747460	0.747720	0.998590	0.999690
Mission Reliability (50% Confidence)	0.748625	0.748883	0.998700	0.999742
Total Rate Per Million Missions	251375	251117	1300	258
COMPONENT				
Hot Gas Manifold	11283	11282	1.2	1.1
Comb. Chamber	95953	95902	52.0	51.2
Main Injector	9496	9484	92.9	12.4
Oxidizer Preburner	3593	3590	31.0	2.7
Fuel Preburner	13909	13908	31.0	1.0
High Press. Fuel Pump	47683	47607	185.9	76.4
High Press.Oxid. Pump	27606	27561	62.0	44.7
Low Press. Fuel Pump	17459	17457	1.5	1.3
Low Press.Oxid. Pump	3073	3068	5.0	5.0
Nozzle	13514	13483	154.9	30.8
Heat Exchanger	6060	6058	2.0	2.3
MCC Ignitor	295	295	0.0	0.0
Fuel Inlet	40	40	13.8	0.1
Oxidizer Inlet	40	40	13.8	0.1
Fuel Flow Cntr.	40	40	13.8	0.1
Oxidizer Flow Cntr.	40	40	13.8	0.1
Fuel Pre-Brn	40	40	13.8	0.1
Oxid. Pre-Brn	40	40	13.8	0.1
Solenoid	40	40	13.8	0.1
H2 Check Valve	40	40	13.8	0.1
O2 Check Valve	40	40	13.8	0.1
Fuel/Hot Gas System	37	24	31.0	13.1
Oxidizer System	37	32	31.0	5.5
Thrust Cntr.	0	0	0.0	0.0
Pneumatic Control Sys.	111	103	92.9	8.5
Controller(Electronics)	74	74	62.0	0.0
Controller(Software)	74	74	62.0	0.0
Control Sensors&Har.	148	148	123.9	0.0
Hydraulic System	284	284	0.0	0.0
Actuators	324	323	154.9	1.2

** BASED ON 100,000 MISSION RUNS IN RELIABILITY MODEL

Engine Level Summary

Component Level Rates

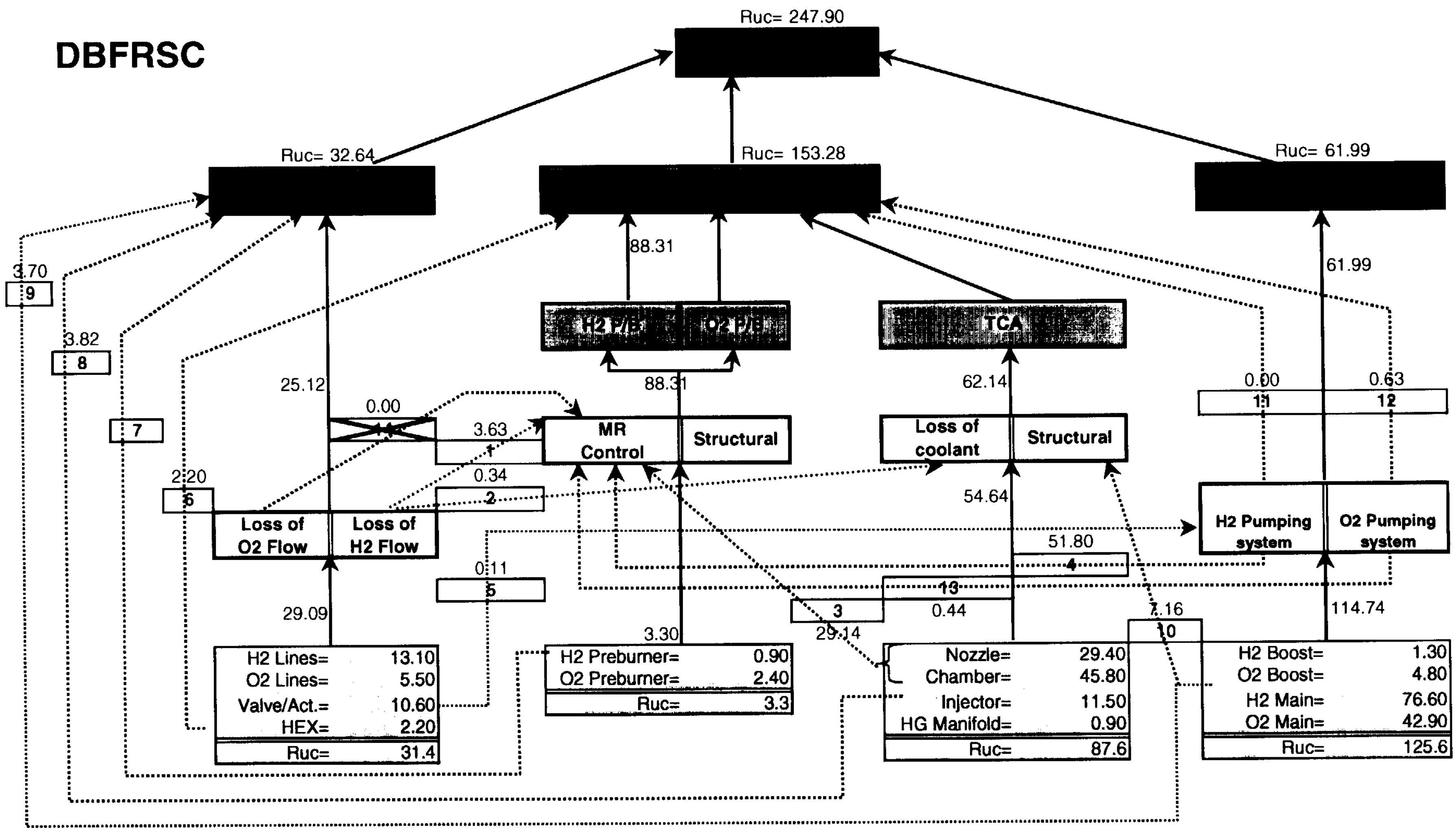
SSME



Ruc= 258.10
[redacted]
Ruc= 33.25
[redacted]
Ruc= 161.15
[redacted]
Ruc= 63.69
[redacted]
91.70
63.69
H2 P/B
O2 P/B
TCA
3.69
9
4.12
8
0.33
7
2.30
6
25.12
0.00
14
3.63
1
0.34
2
0.11
5
91.70
MR Control
Structural
66.50
Loss of coolant
Structural
58.81
0.00
11
0.65
12
Loss of O2 Flow
Loss of H2 Flow
H2 Pumping system
O2 Pumping system
51.66
4
13
0.46
3
32.68
7.35
10
116.36
29.09
3.37
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 2.30
Ruc= 31.5
H2 Preburner= 1.00
O2 Preburner= 2.70
Ruc= 3.7
Nozzle= 30.80
Chamber= 51.20
Injector= 12.40
HG Manifold= 1.10
Ruc= 95.5
H2 Boost= 1.30
O2 Boost= 5.00
H2 Main= 76.40
O2 Main= 44.70
Ruc= 127.4

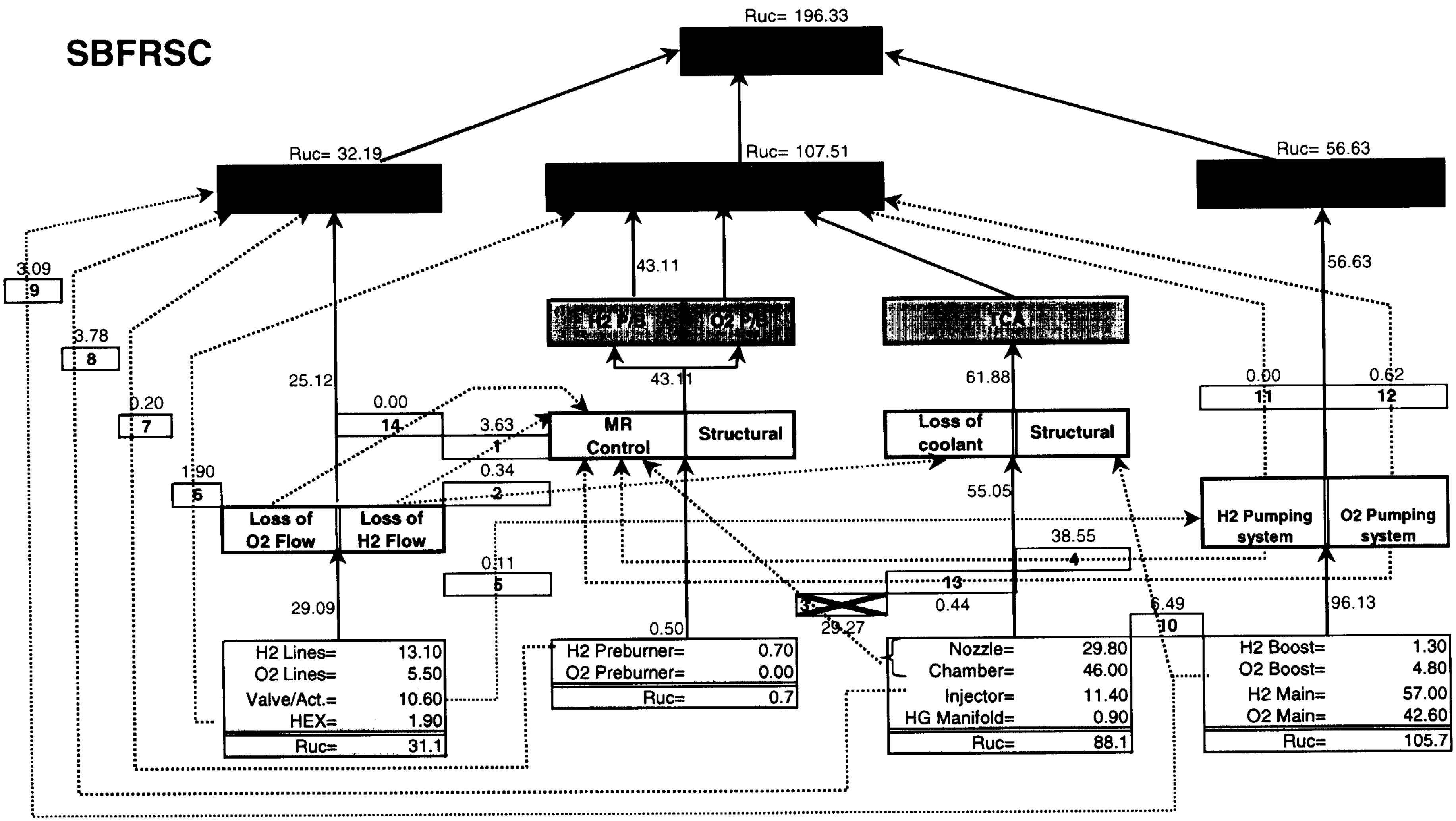
11.0 APPENDIX E – BASELINE CYCLE RELIABILITY

DBFRSC

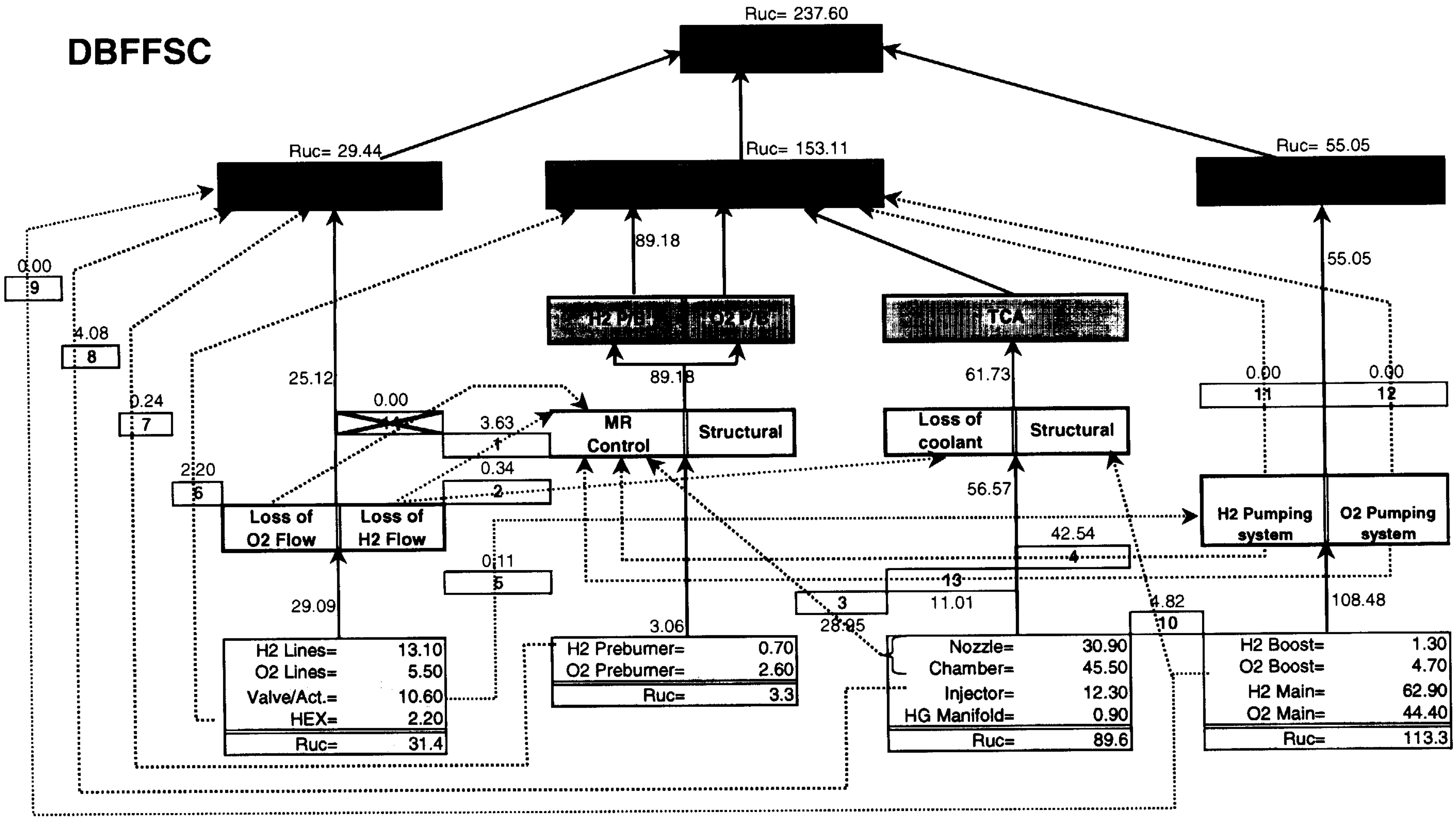


Ruc= 247.90
Ruc= 32.64
Ruc= 153.28
Ruc= 61.99
88.31
61.99
H2 P/B
O2 P/B
TCA
25.12
0.00
3.63
1
0.34
2
88.31
62.14
0.00
0.63
11
12
MR Control
Structural
Loss of coolant
Structural
2.20
6
Loss of O2 Flow
Loss of H2 Flow
54.64
H2 Pumping system
O2 Pumping system
51.80
4
0.11
5
13
3
0.44
7.16
10
114.74
29.09
3.30
29.14
3.70
9
3.82
8
7
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 2.20
Ruc= 31.4
H2 Preburner= 0.90
O2 Preburner= 2.40
Ruc= 3.3
Nozzle= 29.40
Chamber= 45.80
Injector= 11.50
HG Manifold= 0.90
Ruc= 87.6
H2 Boost= 1.30
O2 Boost= 4.80
H2 Main= 76.60
O2 Main= 42.90
Ruc= 125.6

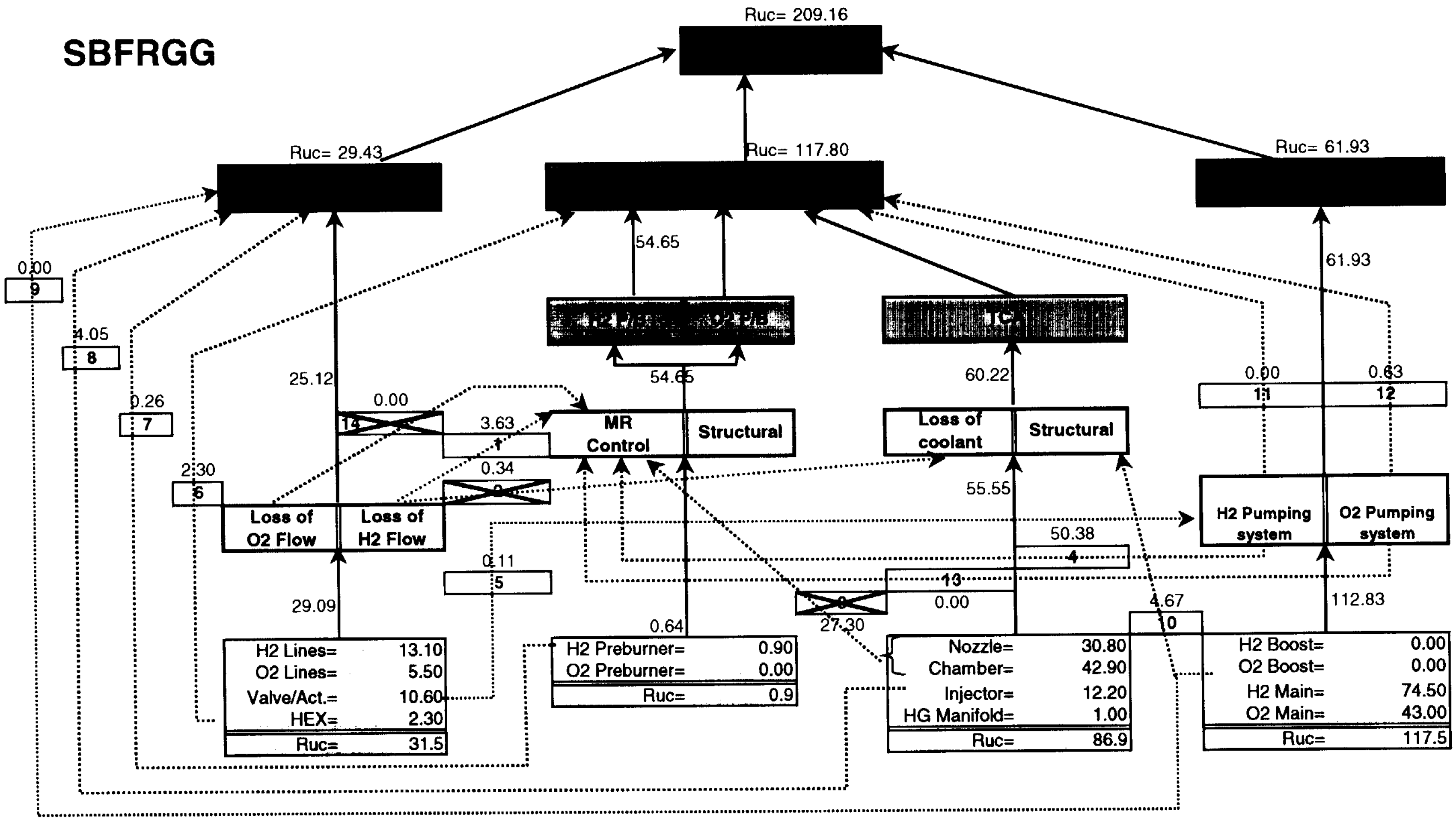
SBFRSC


Ruc= 196.33
[redacted]
Ruc= 32.19
[redacted]
Ruc= 107.51
[redacted]
Ruc= 56.63
[redacted]
43.11
56.63
H2 P/B
O2 P/B
TCA
3.09
9
3.78
8
0.20
7
1.90
6
25.12
0.00
14
3.63
1
0.34
2
43.11
61.88
0.00
11
0.62
12
MR Control
Structural
Loss of coolant
Structural
Loss of O2 Flow
Loss of H2 Flow
55.05
H2 Pumping system
O2 Pumping system
38.55
4
0.11
5
13
3
0.44
29.27
6.49
10
29.09
0.50
96.13
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 1.90
Ruc= 31.1
H2 Preburner= 0.70
O2 Preburner= 0.00
Ruc= 0.7
Nozzle= 29.80
Chamber= 46.00
Injector= 11.40
HG Manifold= 0.90
Ruc= 88.1
H2 Boost= 1.30
O2 Boost= 4.80
H2 Main= 57.00
O2 Main= 42.60
Ruc= 105.7

DBFFSC

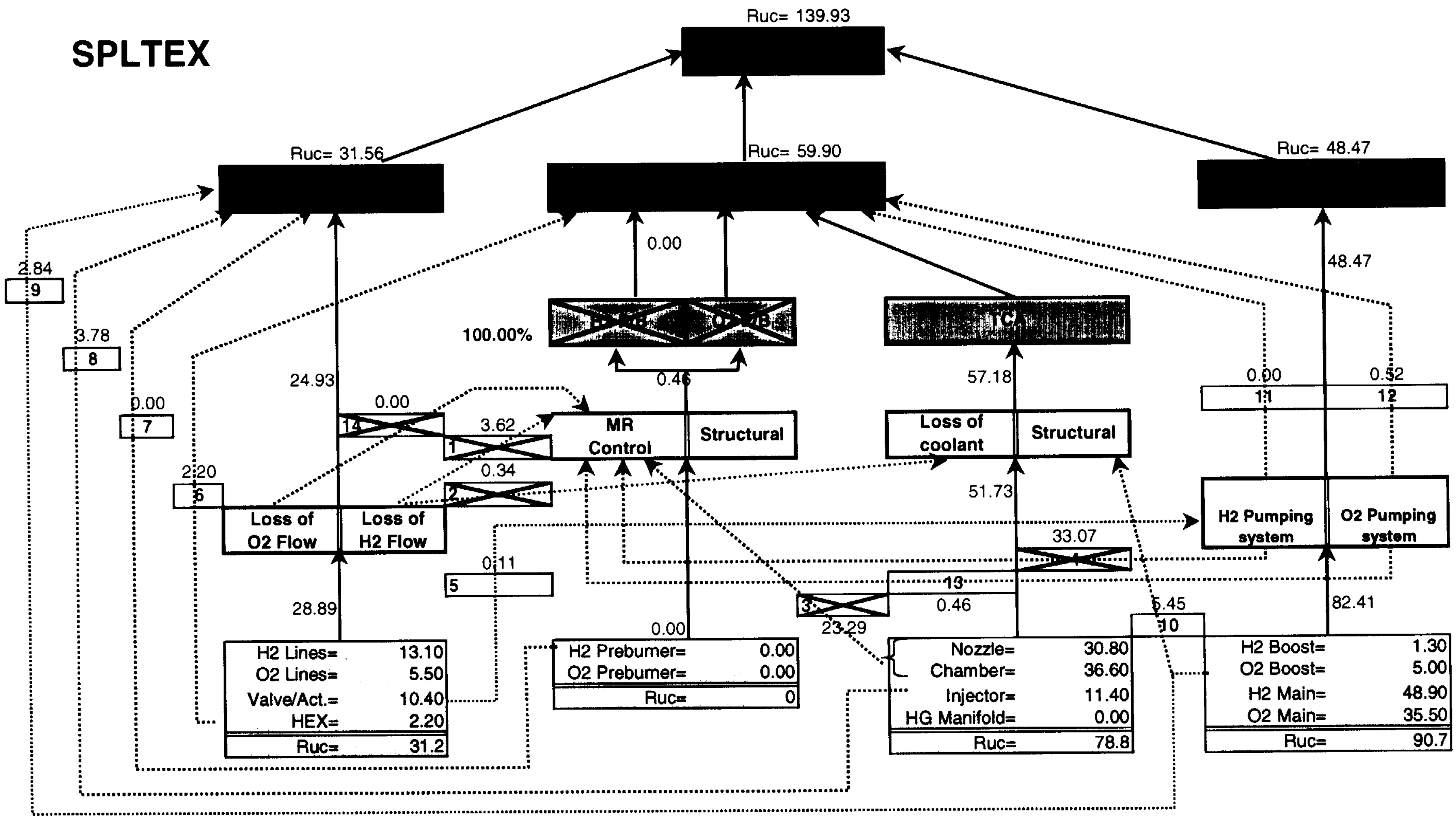

Ruc= 237.60
Ruc= 29.44
Ruc= 153.11
Ruc= 55.05
89.18
55.05
0.00
9
4.08
8
0.24
7
2.20
6
25.12
0.00
14
3.63
1
0.34
2
0.11
5
H2 P/B
O2 P/B
TCA
89.18
61.73
0.00
11
0.00
12
MR Control
Structural
Loss of coolant
Structural
56.57
Loss of O2 Flow
Loss of H2 Flow
H2 Pumping system
O2 Pumping system
42.54
4
13
3
11.01
4.82
10
108.48
29.09
3.06
28.95
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 2.20
Ruc= 31.4
H2 Preburner= 0.70
O2 Preburner= 2.60
Ruc= 3.3
Nozzle= 30.90
Chamber= 45.50
Injector= 12.30
HG Manifold= 0.90
Ruc= 89.6
H2 Boost= 1.30
O2 Boost= 4.70
H2 Main= 62.90
O2 Main= 44.40
Ruc= 113.3

SBFRGG



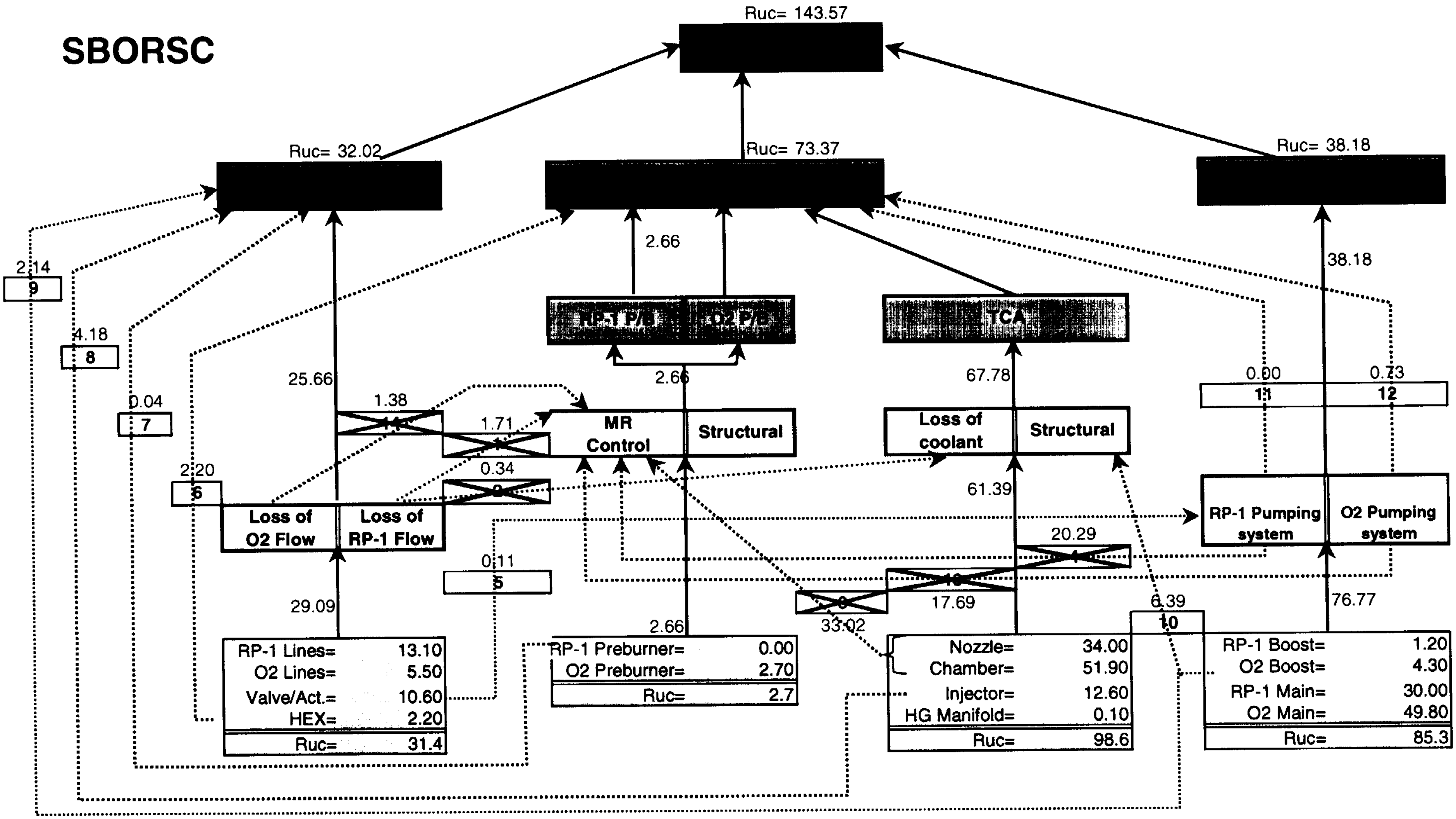
Ruc= 209.16
Ruc= 29.43
Ruc= 117.80
Ruc= 61.93
54.65
61.93
0.00
9
4.05
8
0.26
7
25.12
0.00
14
3.63
1
MR Control
Structural
54.65
60.22
Loss of coolant
Structural
0.00
11
0.63
12
2.30
6
0.34
2
Loss of O2 Flow
Loss of H2 Flow
55.55
H2 Pumping system
O2 Pumping system
50.38
4
0.11
5
13
29.09
0.00
4.67
10
112.83
0.64
27.30
3
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 2.30
Ruc= 31.5
H2 Preburner= 0.90
O2 Preburner= 0.00
Ruc= 0.9
Nozzle= 30.80
Chamber= 42.90
Injector= 12.20
HG Manifold= 1.00
Ruc= 86.9
H2 Boost= 0.00
O2 Boost= 0.00
H2 Main= 74.50
O2 Main= 43.00
Ruc= 117.5

SPLTEX



Ruc= 139.93
[redacted]
Ruc= 31.56
[redacted]
Ruc= 59.90
[redacted]
Ruc= 48.47
[redacted]
0.00
48.47
2.84
9
3.78
8
0.00
7
2.20
6
100.00%
TC
24.93
0.00
14
3.62
1
0.34
2
0.46
MR Control
Structural
57.18
Loss of coolant
Structural
0.00
11
0.52
12
51.73
Loss of O2 Flow
Loss of H2 Flow
H2 Pumping system
O2 Pumping system
33.07
4
0.11
5
13
0.46
3
23.29
5.45
10
82.41
28.89
0.00
H2 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.40
HEX= 2.20
Ruc= 31.2
H2 Prebumer= 0.00
O2 Prebumer= 0.00
Ruc= 0
Nozzle= 30.80
Chamber= 36.60
Injector= 11.40
HG Manifold= 0.00
Ruc= 78.8
H2 Boost= 1.30
O2 Boost= 5.00
H2 Main= 48.90
O2 Main= 35.50
Ruc= 90.7

SBORSC


Ruc= 143.57
Ruc= 32.02
Ruc= 73.37
Ruc= 38.18
2.66
38.18
2.14
9
4.18
8
0.04
7
2.20
6
25.66
1.38
14
1.71
13
0.34
2.66
RP-1 P/B
O2 P/B
TCA
67.78
0.00
11
0.73
12
MR Control
Structural
Loss of coolant
Structural
61.39
Loss of O2 Flow
Loss of RP-1 Flow
RP-1 Pumping system
O2 Pumping system
20.29
0.11
5
29.09
17.69
33.02
6.39
10
76.77
2.66
RP-1 Lines= 13.10
O2 Lines= 5.50
Valve/Act.= 10.60
HEX= 2.20
Ruc= 31.4
RP-1 Preburner= 0.00
O2 Preburner= 2.70
Ruc= 2.7
Nozzle= 34.00
Chamber= 51.90
Injector= 12.60
HG Manifold= 0.10
Ruc= 98.6
RP-1 Boost= 1.20
O2 Boost= 4.30
RP-1 Main= 30.00
O2 Main= 49.80
Ruc= 85.3

Component	Cycle Adjusted QRAS Hardware Failure Rate (R'f) per Million						
	Base Line	DBFRSC	SBFRSC	DBFFSC	SBFRGG	SBORSC	SPLTEX
	251375	234794	215672	226022	206080	201986	167038
Hot Gas Manifold	11283	9371	9089	8851	9415	690	0
Comb. Chamber	95953	85847	86208	85199	80470	97199	68650
Main Injector	9496	8838	8759	9469	9414	9667	8783
Oxidizer Preburner	3593	3199	0	3518	0	3664	0
Fuel Preburner	13909	12305	9378	10785	12433	0	0
High Press. Fuel Pump	47683	47827	35567	39259	46512	18747	30419
High Press. Oxid. Pump	27606	26457	26316	27387	26514	30747	17506
Low Press. Fuel Pump	17459	17369	17369	17459	0	16121	17459
Low Press. Oxid. Pump	3073	2950	2919	2858	0	2643	3073
Nozzle	13514	12915	13101	13588	13514	14938	13514
Heat Exchanger	6060	5968	5217	5902	6060	5823	5968
MCC Ignitor	295	295	295	295	295	295	295
Fuel Inlet	40	40	40	40	40	40	40
Oxidizer Inlet	40	40	40	40	40	40	40
Fuel Flow Cntr.	40	40	40	40	40	40	40
Oxidizer Flow Cntr.	40	40	40	40	40	40	40
Fuel Pre-Brn	40	40	40	40	40	40	0
Oxid. Pre-Brn	40	40	40	40	40	40	0
Solenoid	40	40	40	40	40	40	40
H2 Check Valve	40	40	40	40	40	40	40
O2 Check Valve	40	40	40	40	40	40	40
Fuel/Hot Gas System	37	37	37	37	37	37	37
Oxidizer System	37	37	37	37	37	37	37
Thrust Cntr.	0	0	0	0	0	0	1
Pneumatic Control Sys.	111	111	111	111	111	111	111
Controller(Electronics)	74	74	74	74	74	74	74
Controller(Software)	74	74	74	74	74	74	74
Control Sensors&Har.	148	148	148	148	148	148	148
Hydraulic System	284	284	284	284	284	284	284
Actuators	324	324	324	324	324	324	324

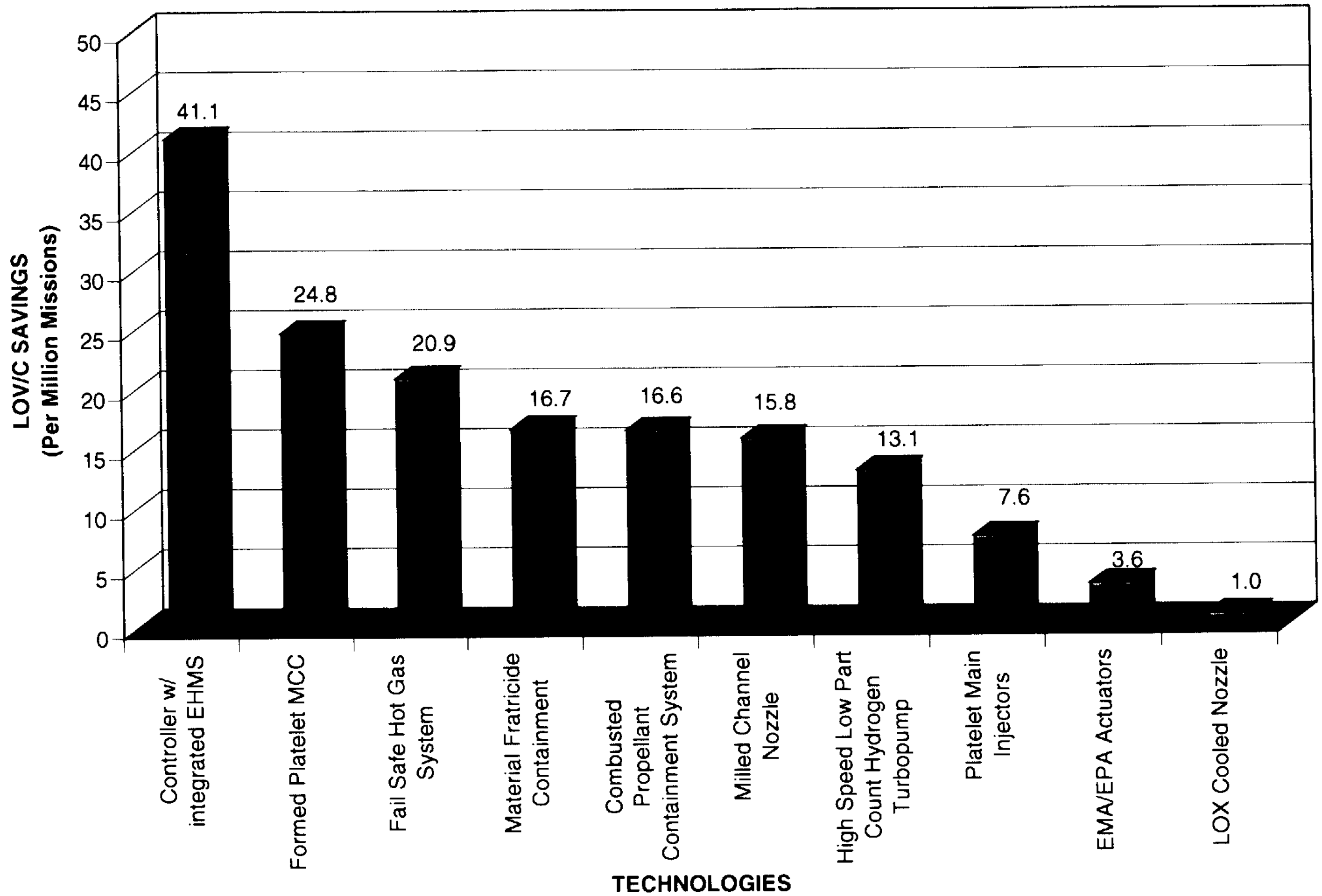
Component	Cycle Adjusted QRAS Shutdown Rate (R's) per Million						
	SSME Baseline	DBFRSC	SBFRSC	DBFFSC	SBFRGG	SBORSC	SPLTEX
	1300.8	1262.8	1181.8	1244.3	1235.8	1170.8	1090.8
Hot Gas Manifold	1.2	0.8	0.8	0.8	0.8	0.1	0.0
Comb. Chamber	52.0	38.4	38.6	38.1	36.0	43.5	30.7
Main Injector	92.9	86.5	85.7	92.7	92.1	94.6	86.0
Oxidizer Preburner	31.0	27.6	0.0	30.3	0.0	31.6	0.0
Fuel Preburner	31.0	27.4	20.9	24.0	27.7	0.0	0.0
High Press. Fuel Pump	185.9	186.4	138.6	153.0	181.3	73.1	118.6
High Press. Oxid. Pump	62.0	59.4	59.1	61.5	59.5	69.0	39.3
Low Press. Fuel Pump	1.5	1.0	1.0	1.0	0.0	0.9	1.0
Low Press. Oxid. Pump	5.0	3.8	3.8	3.7	0.0	3.4	4.0
Nozzle	154.9	148.0	150.2	155.7	154.9	171.2	154.9
Heat Exchanger	2.0	2.0	1.7	1.9	2.0	1.9	2.0
MCC Ignitor	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Fuel Inlet	13.8	13.8	13.8	13.8	13.8	13.8	13.8
Oxidizer Inlet	13.8	13.8	13.8	13.8	13.8	13.8	13.8
Fuel Flow Cntr.	13.8	13.8	13.8	13.8	13.8	13.8	13.8
Oxidizer Flow Cntr.	13.8	13.8	13.8	13.8	13.8	13.8	13.8
Fuel Pre-Brn	13.8	13.8	13.8	13.8	13.8	13.8	0.0
Oxid. Pre-Brn	13.8	13.8	13.8	13.8	13.8	13.8	0.0
Solenoid	13.8	13.8	13.8	13.8	13.8	13.8	13.8
H2 Check Valve	13.8	13.8	13.8	13.8	13.8	13.8	13.8
O2 Check Valve	13.8	13.8	13.8	13.8	13.8	13.8	13.8
Fuel/Hot Gas System	31.0	31.0	31.0	31.0	31.0	31.0	31.0
Oxidizer System	31.0	31.0	31.0	31.0	31.0	31.0	31.0
Thrust Cntr.	0.0	0.0	0.0	0.0	0.0	0.0	0.5
Pneumatic Control Sys.	92.9	92.9	92.9	92.9	92.9	92.9	92.9
Controller(Electronics)	62.0	62.0	62.0	62.0	62.0	62.0	62.0
Controller(Software)	62.0	62.0	62.0	62.0	62.0	62.0	62.0
Control Sensors&Har.	123.9	123.9	123.9	123.9	123.9	123.9	123.9
Hydraulic System	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Actuators	154.9	154.9	154.9	154.9	154.9	154.9	154.9

Component	Cycle Adjusted QRAS Uncontained Rate (R'uc) per Million						
	SSME Baseline	DBFRSC	SBFRSC	DBFFSC	SBFRGG	SBORSC	SPLTEX
	258	248	226	238	237	218	194
Hot Gas Manifold	1.1	0.9	0.9	0.9	1.0	0.1	0.0
Comb. Chamber	51.2	45.8	46.0	45.5	42.9	51.9	36.6
Main Injector	12.4	11.5	11.4	12.3	12.2	12.6	11.4
Oxidizer Preburner	2.7	2.4	0.0	2.6	0.0	2.7	0.0
Fuel Preburner	1.0	0.9	0.7	0.7	0.9	0.0	0.0
High Press. Fuel Pump	76.4	76.6	57.0	62.9	74.5	30.0	48.7
High Press. Oxid. Pump	44.7	42.9	42.6	44.4	43.0	49.8	28.4
Low Press. Fuel Pump	1.3	1.3	1.3	1.3	0.0	1.2	1.3
Low Press. Oxid. Pump	5.0	4.8	4.8	4.7	0.0	4.3	5.0
Nozzle	30.8	29.4	29.8	30.9	30.8	34.0	30.8
Heat Exchanger	2.3	2.2	1.9	2.2	2.3	2.2	2.2
MCC Ignitor	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Fuel Inlet	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Oxidizer Inlet	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Fuel Flow Cntr.	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Oxidizer Flow Cntr.	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Fuel Pre-Brn	0.1	0.1	0.1	0.1	0.1	0.1	0.0
Oxid. Pre-Brn	0.1	0.1	0.1	0.1	0.1	0.1	0.0
Solenoid	0.1	0.1	0.1	0.1	0.1	0.1	0.1
H2 Check Valve	0.1	0.1	0.1	0.1	0.1	0.1	0.1
02 Check Valve	0.1	0.1	0.1	0.1	0.1	0.1	0.1
Fuel/Hot Gas System	13.1	13.1	13.1	13.1	13.1	13.1	13.1
Oxidizer System	5.5	5.5	5.5	5.5	5.5	5.5	5.5
Thrust Cntr.	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Pneumatic Control Sys.	8.5	8.5	8.5	8.5	8.5	8.5	8.5
Controller(Electronics)	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Controller(Software)	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Control Sensors&Har.	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Hydraulic System	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Actuators	1.2	1.2	1.2	1.2	1.2	1.2	1.2

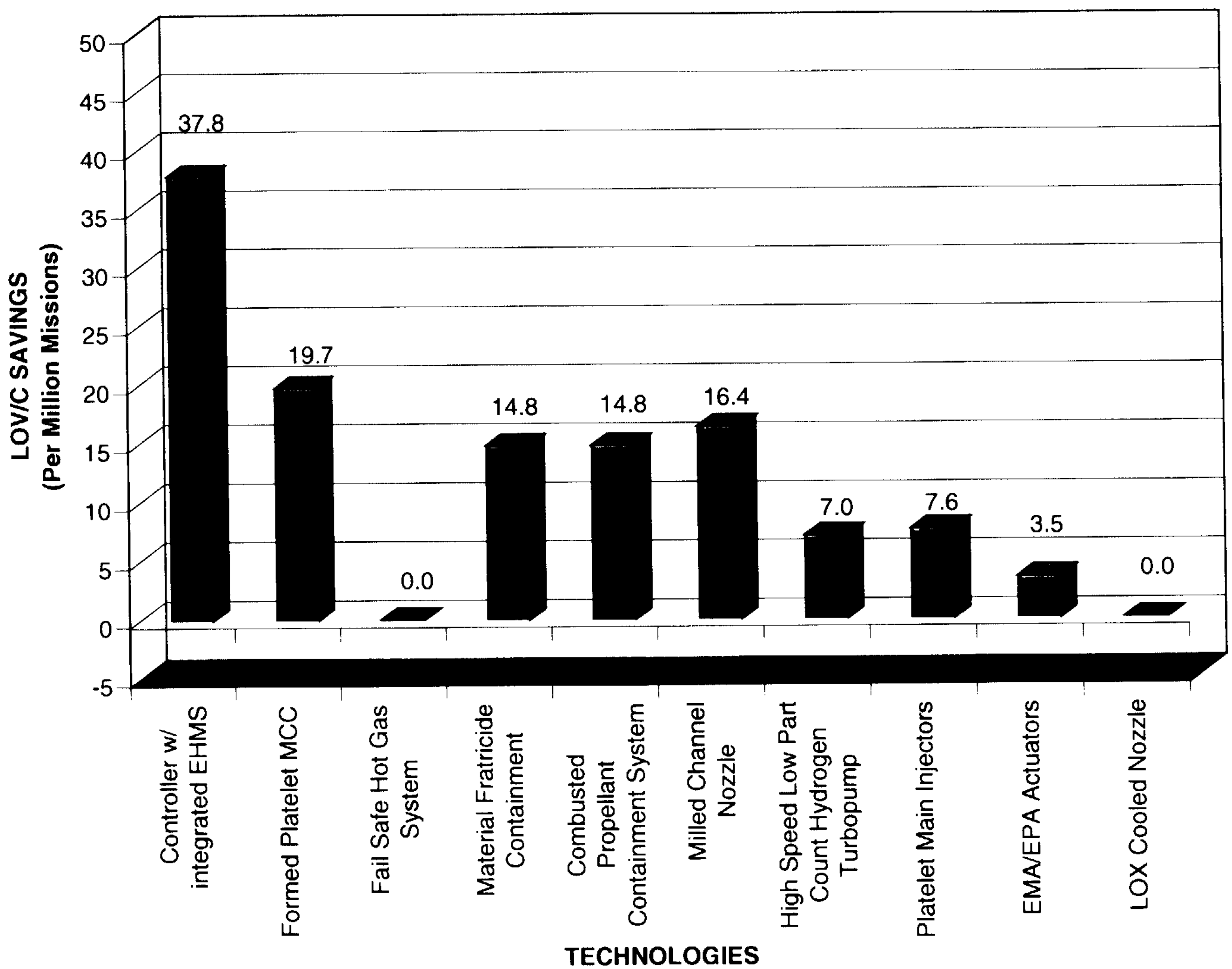
* This data was used at the input for the failure propagation trees

12.0 APPENDIX F – IMPROVED CYCLE RELIABILITY

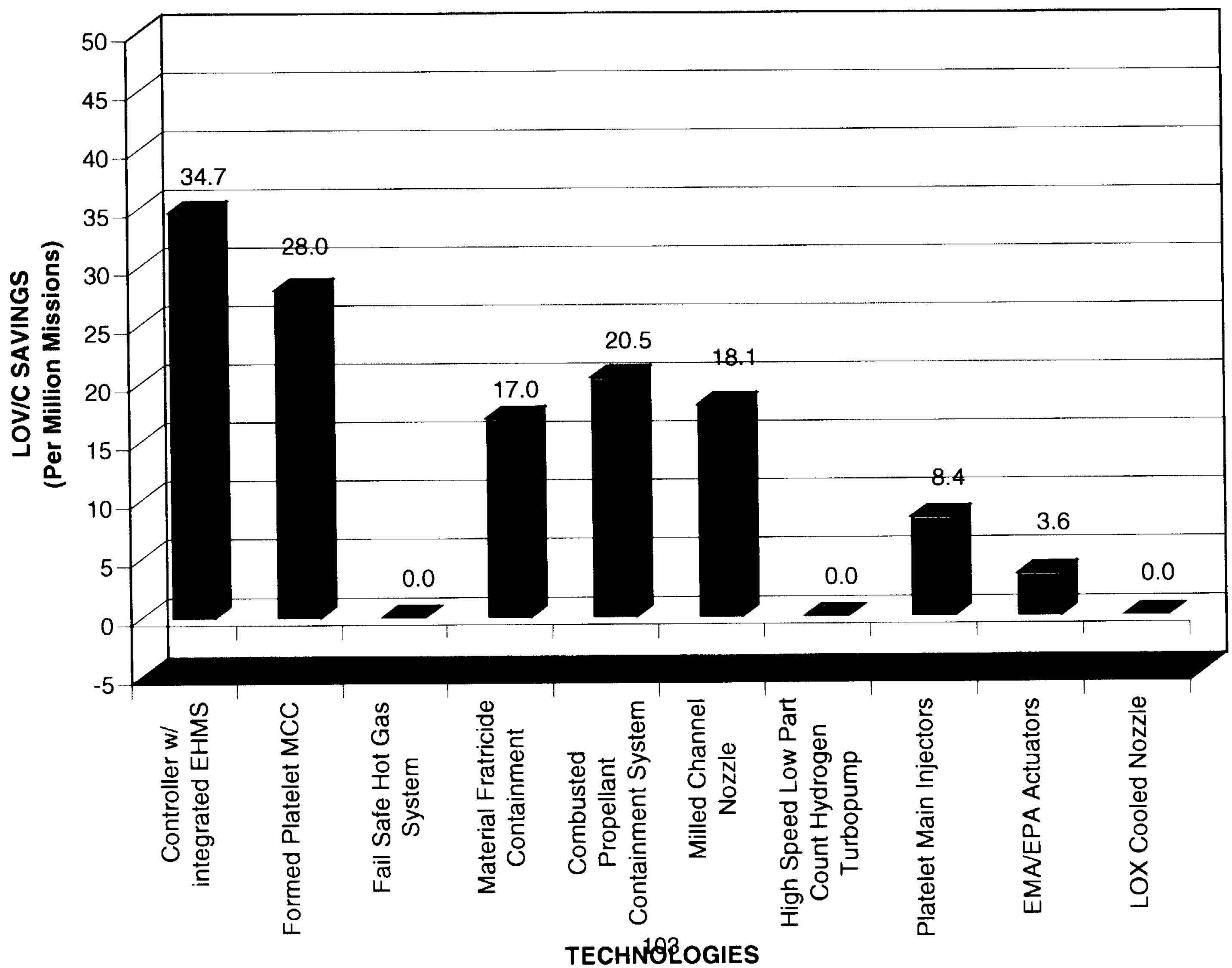
SBFRSC TECHNOLOGY LOV/C SAVINGS

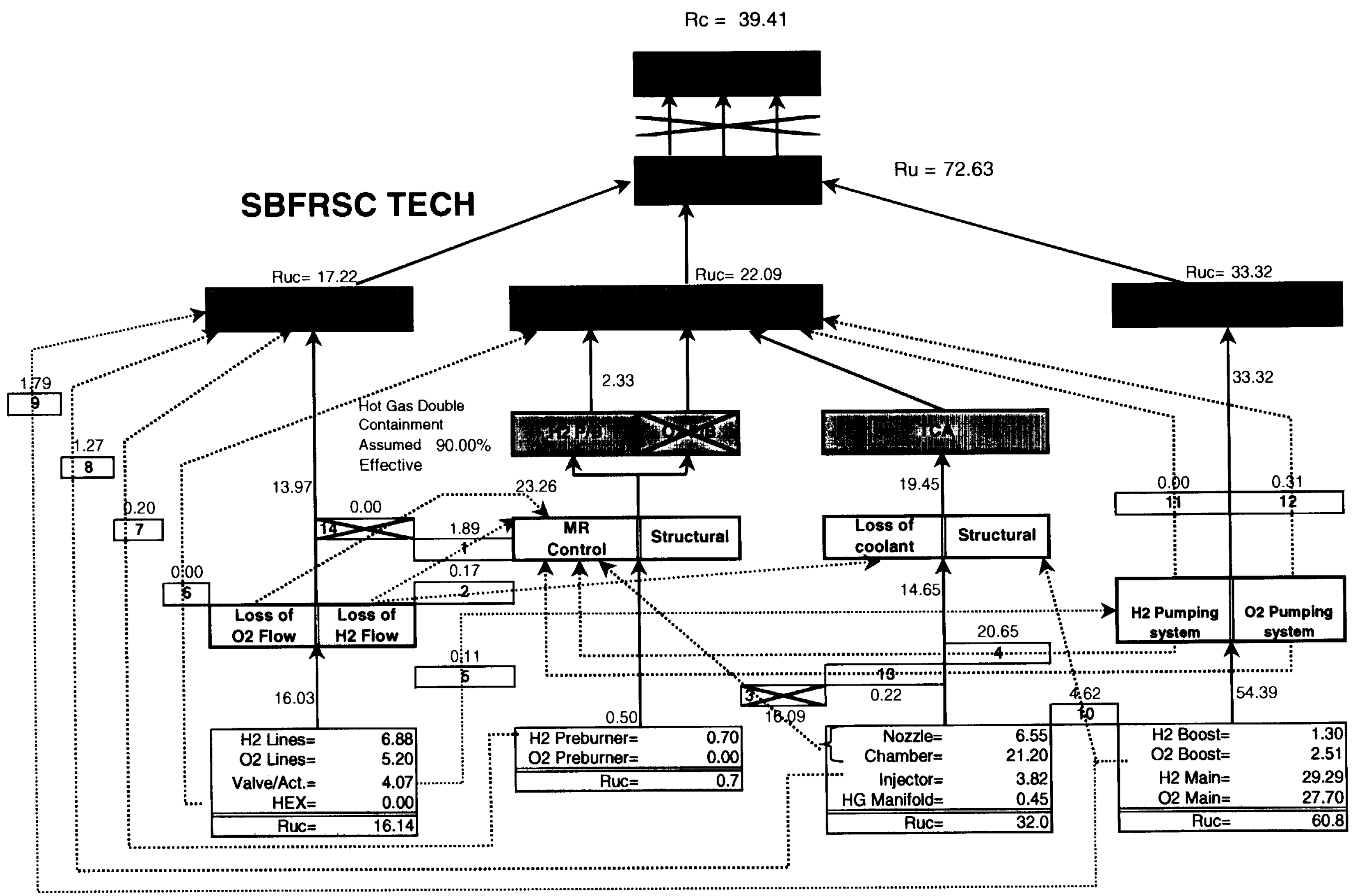


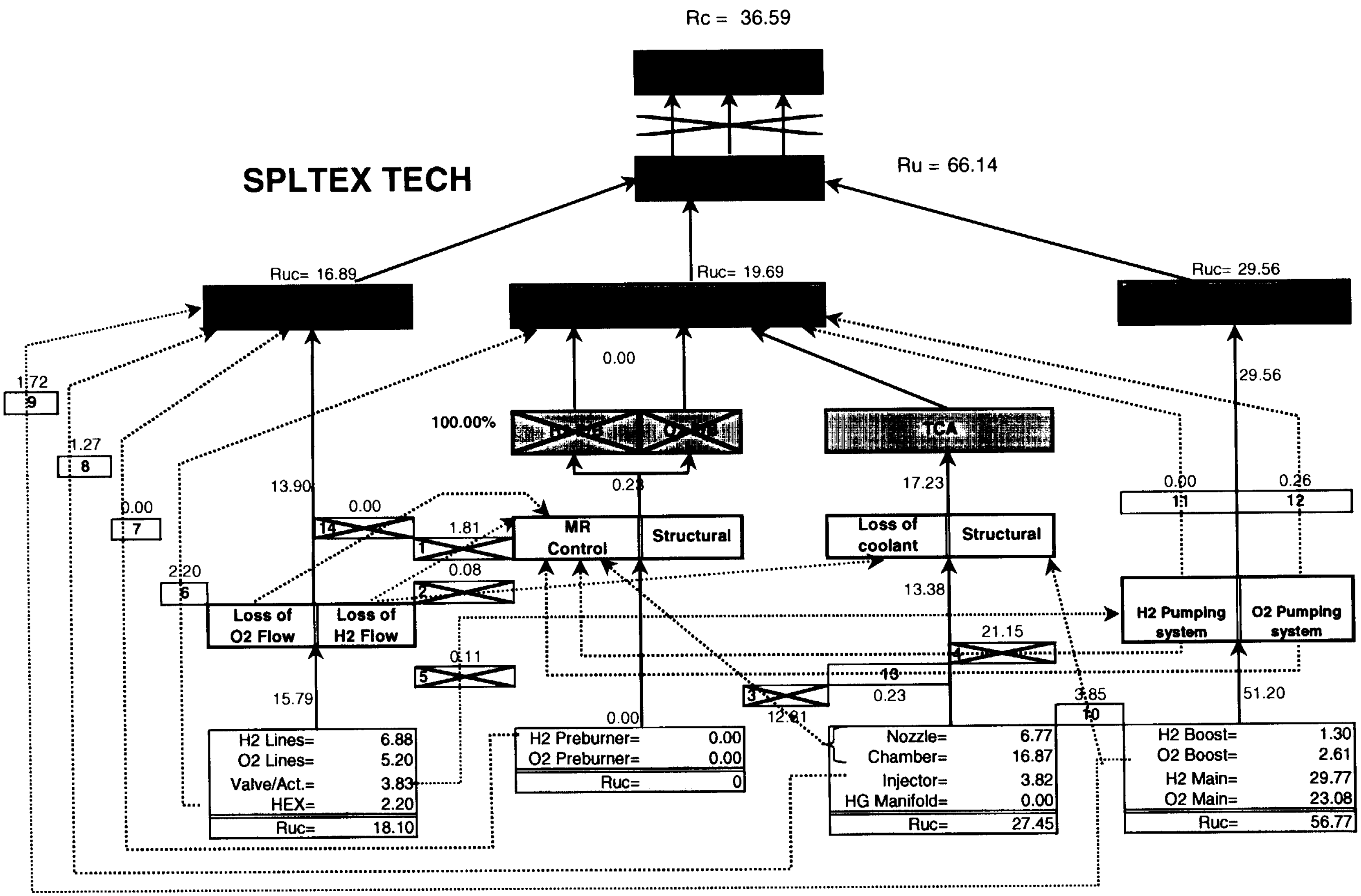
SPLTEX TECHNOLOGY LOV/C SAVINGS



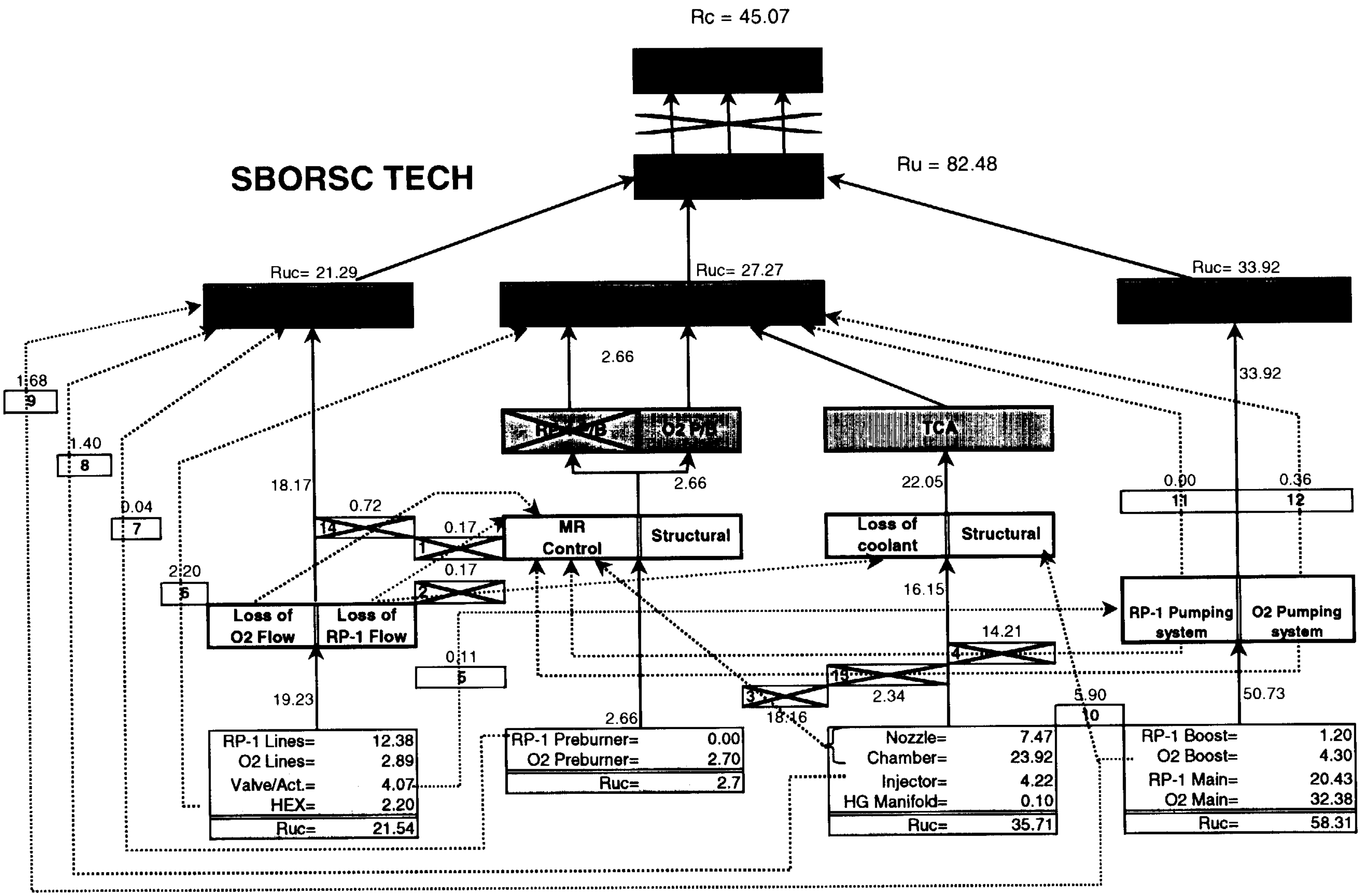
SBORSC TECHNOLOGY LOV/C SAVINGS




Rc = 39.41
Ru = 72.63
SBFRSC TECH
Ruc= 17.22
Ruc= 22.09
Ruc= 33.32
1.79
9
1.27
8
0.20
7
0.00
6
Hot Gas Double Containment Assumed 90.00% Effective
13.97
0.00
14
1.89
1
0.17
2
0.11
5
23.26
2.33
TCA
19.45
33.32
0.00
11
0.31
12
MR Control
Structural
Loss of coolant
Structural
Loss of O2 Flow
Loss of H2 Flow
H2 Pumping system
O2 Pumping system
14.65
20.65
4
13
0.22
3
18.09
4.62
10
54.39
16.03
0.50
H2 Lines= 6.88
O2 Lines= 5.20
Valve/Act.= 4.07
HEX= 0.00
Ruc= 16.14
H2 Preburner= 0.70
O2 Preburner= 0.00
Ruc= 0.7
Nozzle= 6.55
Chamber= 21.20
Injector= 3.82
HG Manifold= 0.45
Ruc= 32.0
H2 Boost= 1.30
O2 Boost= 2.51
H2 Main= 29.29
O2 Main= 27.70
Ruc= 60.8

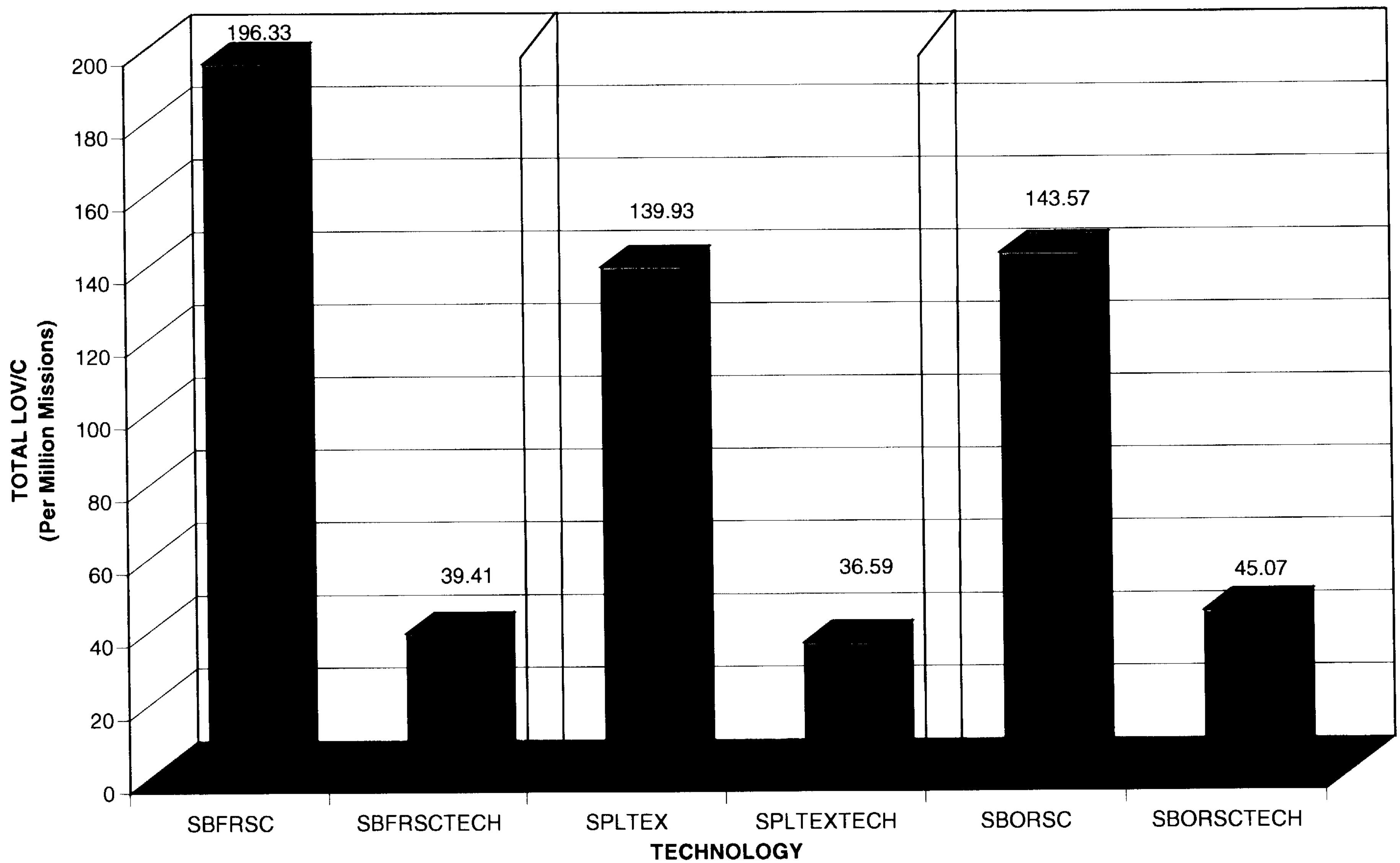

Rc = 36.59
SPLTEX TECH
Ru = 66.14
Ruc= 16.89
Ruc= 19.69
Ruc= 29.56
0.00
29.56
100.00%
TCA
1.72
9
1.27
8
0.00
7
2.20
6
13.90
0.00
14
1.81
1
0.23
MR Control
Structural
17.23
Loss of coolant
Structural
0.00
11
0.26
12
0.08
2
13.38
Loss of O2 Flow
Loss of H2 Flow
H2 Pumping system
O2 Pumping system
21.15
4
0.11
5
13
0.23
3
3.85
10
51.20
15.79
0.00
12.81
H2 Lines= 6.88
O2 Lines= 5.20
Valve/Act.= 3.83
HEX= 2.20
Ruc= 18.10
H2 Preburner= 0.00
O2 Preburner= 0.00
Ruc= 0
Nozzle= 6.77
Chamber= 16.87
Injector= 3.82
HG Manifold= 0.00
Ruc= 27.45
H2 Boost= 1.30
O2 Boost= 2.61
H2 Main= 29.77
O2 Main= 23.08
Ruc= 56.77

SBORSC TECH



Rc = 45.07
Ru = 82.48
Ruc= 21.29
Ruc= 27.27
Ruc= 33.92
2.66
33.92
O2 P/B
TCA
1.68
9
1.40
8
0.04
7
2.20
6
18.17
0.72
14
0.17
1
0.17
2
2.66
22.05
0.00
11
0.36
12
MR Control
Structural
Loss of coolant
Structural
Loss of O2 Flow
Loss of RP-1 Flow
16.15
RP-1 Pumping system
O2 Pumping system
14.21
4
0.11
5
15
2.34
3
18.16
5.90
10
50.73
19.23
2.66
RP-1 Lines= 12.38
O2 Lines= 2.89
Valve/Act.= 4.07
HEX= 2.20
Ruc= 21.54
RP-1 Preburner= 0.00
O2 Preburner= 2.70
Ruc= 2.7
Nozzle= 7.47
Chamber= 23.92
Injector= 4.22
HG Manifold= 0.10
Ruc= 35.71
RP-1 Boost= 1.20
O2 Boost= 4.30
RP-1 Main= 20.43
O2 Main= 32.38
Ruc= 58.31

CONSIDERABLE RELIABILITY GAINS CAN BE ACHIEVED WITH TECHNOLOGIES



Component	Shutdown Rate (R"s) per Million			
	BASELINE SSME	SBFRSC	SPLTEX	SBORSC
	1300.8	794.9	823.2	751.6
Hot Gas Manifold	1.2	0.9	0.00	0.02
Comb. Chamber	52.0	27.7	22.92	31.16
Main Injector	92.9	19.5	19.40	23.26
Oxidizer Preburner	31.0	0.0	0.00	11.04
Fuel Preburner	31.0	9.1	0.00	0.00
High Press. Fuel Pump	185.9	94.7	115.00	29.79
High Press. Oxid. Pump	62.0	45.6	38.62	53.22
Low Press. Fuel Pump	1.5	1.5	1.50	1.38
Low Press. Oxid. Pump	5.0	4.8	5.00	4.30
Nozzle	154.9	39.6	41.16	43.46
Heat Exchanger	2.0	0.2	0.20	0.19
MCC Ignitor	0.0	0.0	0.00	0.00
Fuel Inlet	13.8	13.3	13.35	13.35
Oxidizer Inlet	13.8	13.8	13.79	13.79
Fuel Flow Cntr.	13.8	13.8	13.79	13.79
Oxidizer Flow Cntr.	13.8	10.8	10.78	10.78
Fuel Pre-Brn	13.8	6.9	0.00	6.00
Oxid. Pre-Brn	13.8	6.9	0.00	7.00
Solenoid	13.8	13.8	13.79	13.79
H2 Check Valve	13.8	13.8	13.79	13.79
O2 Check Valve	13.8	13.8	13.79	13.79
Fuel/Hot Gas System	31.0	32.6	32.59	38.15
Oxidizer System	31.0	32.6	32.62	30.28
Thrust Cntr.	0.0	0.0	41.88	0.00
Pneumatic Control Sys.	92.9	92.9	92.9	92.9
Controller(Electronics)	62.0	61.9	61.91	61.91
Controller(Software)	62.0	61.9	61.91	61.91
Control Sensors&Har.	123.9	123.7	123.66	123.66
Hydraulic System	0.0	0.0	0.00	0.00
Actuators	154.9	38.9	38.81	38.84

Component	Uncontained Rate (R"uc) per Million		
	SBFRSC	SPLTEX	SBORSC
	39.41	36.61	45.08
Hot Gas Manifold	0.11	0.00	0.03
Comb. Chamber	1.28	1.02	1.44
Main Injector	1.91	1.91	2.11
Oxidizer Preburner	0.00	0.00	0.71
Fuel Preburner	0.21	0.00	0.00
High Press. Fuel Pump	5.07	4.51	3.22
High Press. Oxid. Pump	12.62	10.51	14.75
Low Press. Fuel Pump	1.30	1.30	1.20
Low Press. Oxid. Pump	1.16	1.20	0.99
Nozzle	1.64	1.69	1.87
Heat Exchanger	0.00	0.55	0.55
MCC Ignitor	0.00	0.00	0.00
Fuel Inlet	0.05	0.05	0.05
Oxidizer Inlet	0.05	0.05	0.05
Fuel Flow Cntr.	0.05	0.05	0.05
Oxidizer Flow Cntr.	0.05	0.05	0.05
Fuel Pre-Brn	0.05	0.05	0.05
Oxid. Pre-Brn	0.05	0.05	0.05
Solenoid	0.05	0.05	0.05
H2 Check Valve	0.05	0.05	0.05
O2 Check Valve	0.05	0.05	0.05
Fuel/Hot Gas System	5.20	5.16	12.38
Oxidizer System	5.20	5.20	2.17
Thrust Cntr.	0.00	0.05	0.00
Pneumatic Control Sys.	2.97	2.92	2.97
Controller(Electronics)	0.00	0.00	0.00
Controller(Software)	0.00	0.00	0.00
Control Sensors&Har.	0.00	0.00	0.00
Hydraulic System	0.00	0.00	0.00
Actuators	0.28	0.19	0.25

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This is a revised final report and addresses all of the work performed on this program. Specifically, it covers vehicle architecture background, definition of six baseline engine cycles, reliability baseline (space shuttle main engine QRAS), and component level reliability/performance/cost for the six baseline cycles, and selection of 3 cycles for further study. This report further addresses technology improvement selection and component level reliability/performance/cost for the three cycles selected for further study, as well as risk reduction plans, and recommendation for future studies.

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