Spaceliner 100 Candidate Technology
White Paper

Title: Long Life, High Thrust/Weight Lox/Hydrogen Engine.

> This technology assumes the availability of the Propulsion and System IVHM Technologies in order to meet Spaceliner 100 operating cost goals. (This is true of all propulsion technologies for Spaceliner 100).

Technology Category: Flight Systems.

Summary Description: Current reusable O_2/H_2 rocket engines have specific impulse performance almost at the theoretical limit for constant pressure combustion at any given chamber pressure and area ratio. However, they do not have as high a thrust/weight as could be possible, and they require more maintenance and have a shorter life than would be desirable.

This technology, the Long Life, High T/W O_2/H_2 Rocket Engine, specifically addresses the known deficiencies of existing reusable rocket engines and applies the knowledge gained in the 25 years since such an engine was last designed to produce a modern long life, high thrust to weight (~90), low maintenance, reusable O_2/H_2 rocket engine.

This rocket engine needs to be one generation beyond those considered for the RLV and use a power cycle, such as the full flow staged combustion cycle, which specifically allows less severe internal pressures, temperatures, and stresses; wider component trade space; and more component margins and thus lower maintenance and longer life.

As will be shown below, this propulsion technology is a viable candidate for the Spaceliner 100 application and is far less risk than any other propulsion technology being considered.

This approach could be used for engines at most thrust levels. However, one application has been studied in some depth and will be used as an example throughout the rest of this paper. The engine was aimed at a single-stage-to-orbit (SSTO) application. Consequently, it is a high pressure (Pc = 4,000 psi), moderate thrust (421,000 lbf sea level), high mixture ratio (MR = 6.9), bell nozzle engine with an area ratio, sized by its vertical takeoff usage and chamber pressure, of 70.

Engine Power Cycle

To achieve high performance O_2/H_2 engines, chamber pressures in the 3,000 to 5,000 psi range are desired. Also, because high specific impulse performance is wanted, the engines use closed cycles instead of gas generator cycles.

In a closed cycle all of the propellants, even those used to power the turbopumps, flow through the main combustion chamber. Thus all the propellants are used in the manner that extracts the most specific impulse. In contrast, in an open cycle such as a gas generator cycle the propellants used to power the turbopumps are expanded through an auxilary nozzle (or dumped into the main nozzle at a pressure much lower than the chamber pressure) and produce a much lower specific impulse than the rest of the propellants that flow through the main combustion chamber. The effect is that the overall specific impulse is decreased significantly compared to a closed cycle. This effect becomes more severe as the design chamber pressure is increased.

In a closed cycle the amount of energy available to pump the propellants, and thus either increase chamber pressure and engine specific impulse or lower turbine temperatures and pump discharge pressures, is dependent on the regenerative heat from cooling, how much of each propellant is available to the turbine, and whether chemical energy (i.e., preburners) are used to increase the energy of the turbine flows.

The current state of the art is to use a fuel rich staged combustion cycle (SCC), such as that used in the Space Shuttle Main Engine (SSME), to extract energy for high performance engines. This cycle uses preburners to increase the available energy for both the fuel and the oxidizer turbines. However, although all of the fuel flow is used in the preburners, only a small part of the oxidizer flow is used.

More energy is available from the same flow if a different cycle is used. This cycle is called a full flow staged combustion cycle (FFSC). This cycle also uses preburners to increase the available energy for both the fuel and the oxidizer turbines. But in this case all of both the fuel and the oxidizer is used in the preburners. It uses individual preburners to power the fuel turbine and the LOX turbine. The fuel preburner is fuel rich and the LOX preburner is LOX rich. Thus potentially all of both the fuel and LOX flows are available. This cycle can extract the most energy possible for pumping and thus is capable of the highest chamber pressure or of using the lowest turbine temperatures for a given chamber pressure.

In essence using this cycle buys power margin. This can then be used throughout the design, development and fielded phases to increase the trade space available to solve development problems, increase engine life, lower maintenance requirements, de-rate the engine, add thrust, etc. An additional advantage is that this cycle eliminates interpropellant seals in the turbopumps and thus improves safety.

In the Space Shuttle Main Engine (SSME) the chief contributor to life limitations is the internal thermal environment. For example, studies have shown that 70% of the problems encountered in the development and operation of the SSME high pressure fuel turbopump (HPFTP) were thermally induced by the temperatures needed to power the engine, approximately 1,900 R. An evaluation of the impact of turbine operating temperatures for these two cycles (FFSC, SCC) is presented in Figures 1 and 2.

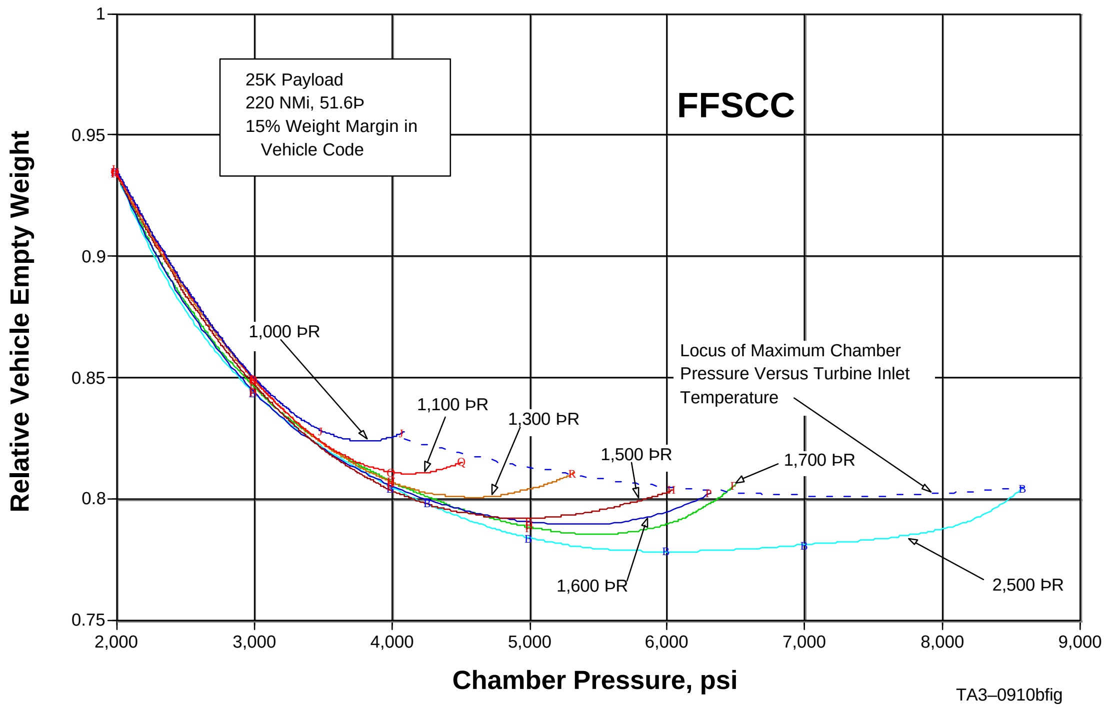

Figure 1. Effect of Fuel Turbine Inlet Temperature – FFSCC

Figure 1 shows the details of the vehicle performance for the FFSC at different fuel turbine operating temperatures. All cases are run with the oxidizer turbine temperature less than or equal to the fuel turbine temperature but high enough that the fuel side is the power limit. For each point, the engine weight is minimized. As the turbine temperature is increased for a given chamber pressure the weight minimization forces the turbine pressure ratio to decrease causing a decrease in pump discharge pressure, a reduction in horsepower and a small decrease in turbopump weight. The pump speeds generally only increase a small amount because the pumps remain at one of three limits (tip speed, hub/tip ratio, or inlet/outlet diameter) and because the turbopumps are pump limited not turbine limited. Consequently, the effect of turbine temperature is minimal except just before the power limit is reached for each temperature.

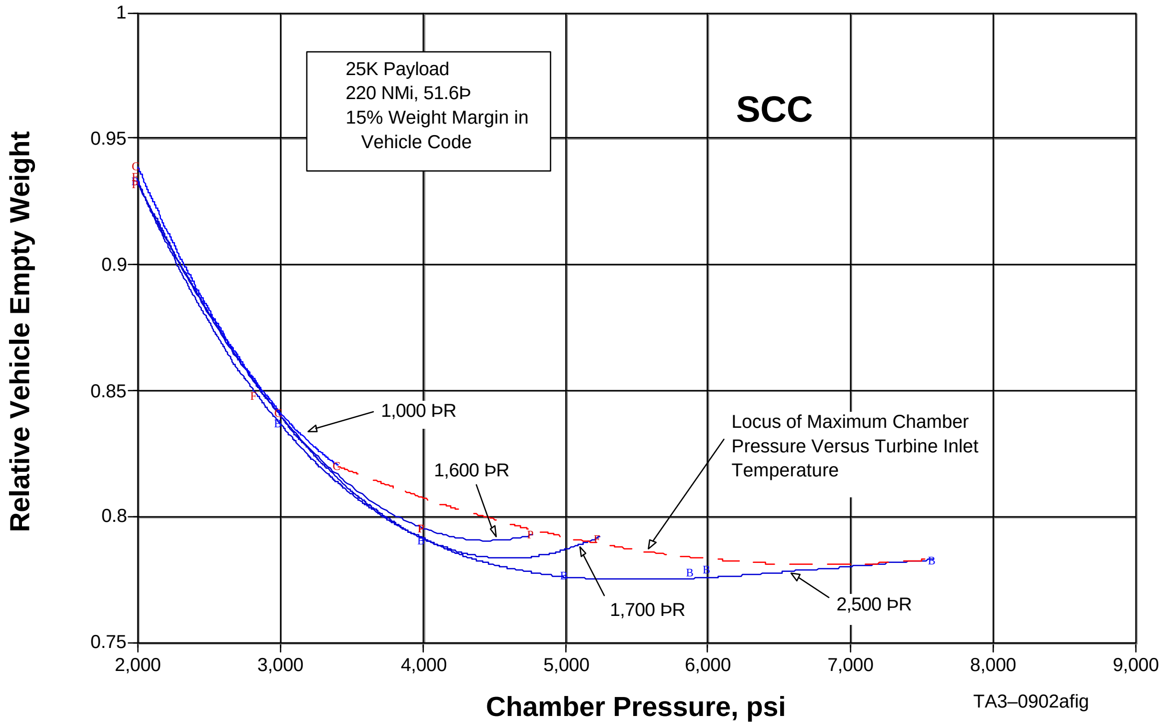

Figure 2. Effect of Fuel Turbine Inlet Temperature – SCC

For closed cycles, there is a sharp increase in engine weight, and thus in the dry vehicle weight seen in Figures 1 and 2 as each line of constant turbine temperature nears its power limit. This occurs because, although the horsepower generated by the turbine is a non-linear function of turbine pressure ratio (increasing rapidly at first, then more and more slowly), the horsepower increase needed in the driven pump is a linear function of turbine pressure ratio. Thus for a given turbine temperature, above a certain turbine pressure ratio, drawing more power from the flow causes a sharp increase in horsepower and therefore turbopump weight. Since the engine weight is minimized for each chamber pressure and high turbine pressure ratio is the parameter which most increases weight, high turbine pressure ratio is the “knob” the minimization procedure turns only when all other parameters have been used. The net effect is that as the power limit is reached the dry vehicle weight line for a given turbine temperature diverges sharply from the lines of higher turbine temperature as is clearly seen in Figures 1 and 2 for the cycles.

At chamber pressures of 3,000 - 5,000 psi, the FFSC cycle can be operated with temperatures in the 1,000 to 1,400 R range with virtually no impact on overall vehicle performance; the SCC can be operated with temperatures in the 1,000 to 1,700 R range. Both cycles can operate at similar temperatures at 3,000 psi but they show significantly different sensitivities to increased chamber pressure. This implies they have different sensitivities to other margins (as would be expected - they have different amounts of energy available). Consequently, the effect of margins was examined.

Engine Margins

Since SSTO engine design is highly driven by performance there is the potential to select design parameters at physical limits (e.g., tip speeds) which produce a point design which is not robust either due to requirements growth or due to component performance sensitivities. Examples include the historical growth of engine thrust requirement and the inability to project specific component performance levels over the development life of an engine. An approach to mitigating these effects and improving engine margin is to design margins into the engine and operate it normally in a de-rated, off-design mode. The issue is the penalty incurred, particularly for the performance-hungry SSTO mission.

Baseline engine configurations were chosen in order to perform margin studies. A chamber pressure of 4,000 psi was chosen and turbine temperatures were chosen as low as possible with the constraint of avoiding condensation in the preburner exhaust. Figure 3 shows the resulting configurations for the cycles.

	FFSCC	SCC
Chamber Pressure, psi	4,000	4,000
Eng Weight, lbm	5,003	4,814
Fuel Tur Temp, ÞR	1,100	1,200
Fuel Dis Press, psi	10,670	11,677
Ox Tur Temp, ÞR	1,100	1,100
Ox Dis Press, psi	9,592	10,984
SSTO Dry Weight, lbm	162,190	159,264

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Figure 3. Baseline Engines for Margin Studies

Five specific margins and their impact on a nominal 4,000 psi chamber pressure, 421 Klb(sl) thrust engine were examined. The five factors were: 5% thrust growth, 5% decrease in all turbomachinery efficiencies, 5:1 throttling capability, increased main combustion chamber (MCC) pressure drop (reflecting a change in cooling channel design and construction technology), increased system pressure drop of 5%, and all of the margins taken together.

Figure 4 shows the effects on fuel turbine operating temperature of each margin and of all the margins together.

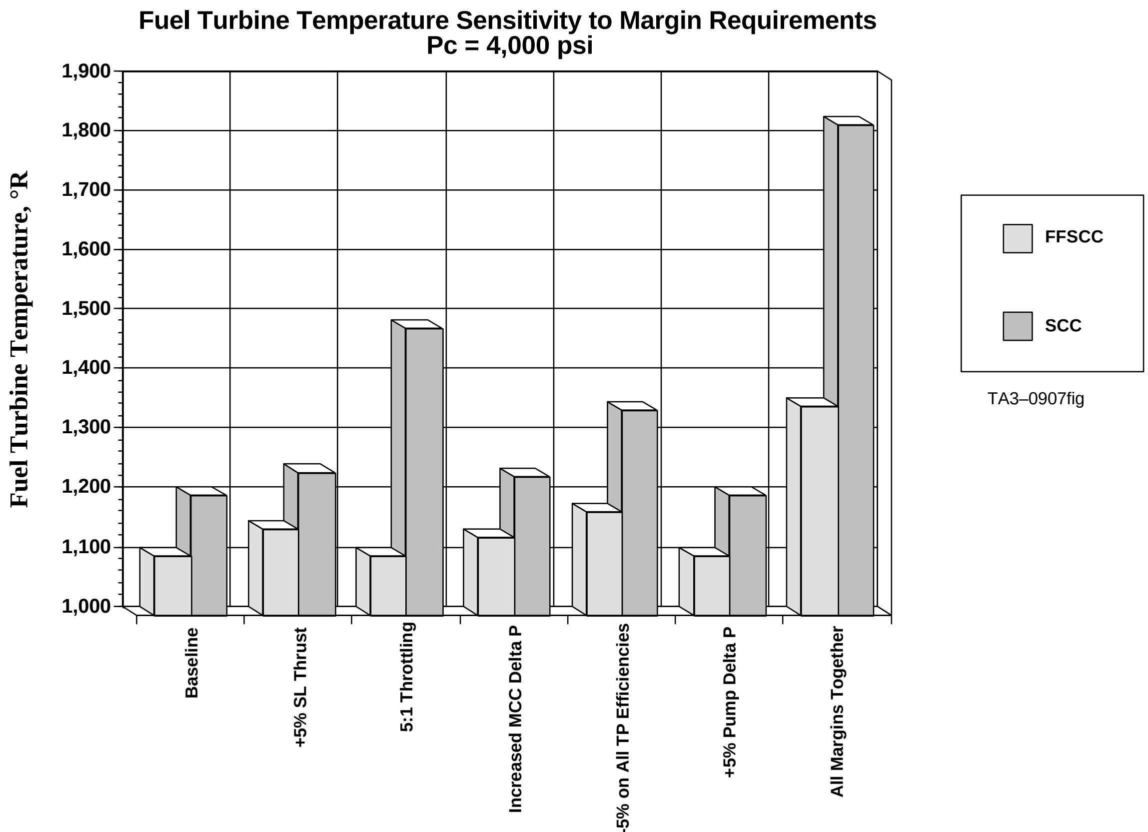

Figure 4. Fuel Turbine Temperature Sensitivity to Margin Requirements

The individual margins have little effect on the FFSC cycle. Even all the margins together leave the turbine temperature at a comfortable level from an engine life point of view. The SCC shows significant sensitivities to the margin requirements and requires life limiting turbine temperatures when the margins are combined. The FFSC was the most robust to the changes in design requirements as would be expected because it has the most power margin.

Figure 5 shows graphically the difference in margins between the two cycles. The use of the FFSC cycle allows qualitative differences in design approach, not just quantitative differences. Cooled hardware can be avoided, reducing complexity, weight, and cost. Materials can operate on the flat portion of their properties versus temperature curves which reduces the safety factors needed to account for engine to engine variations and variations due to component wear over time. In turn, this can greatly improve design predictability and decrease design variability (criteria 15 (number 7 of the asterisked criteria)).

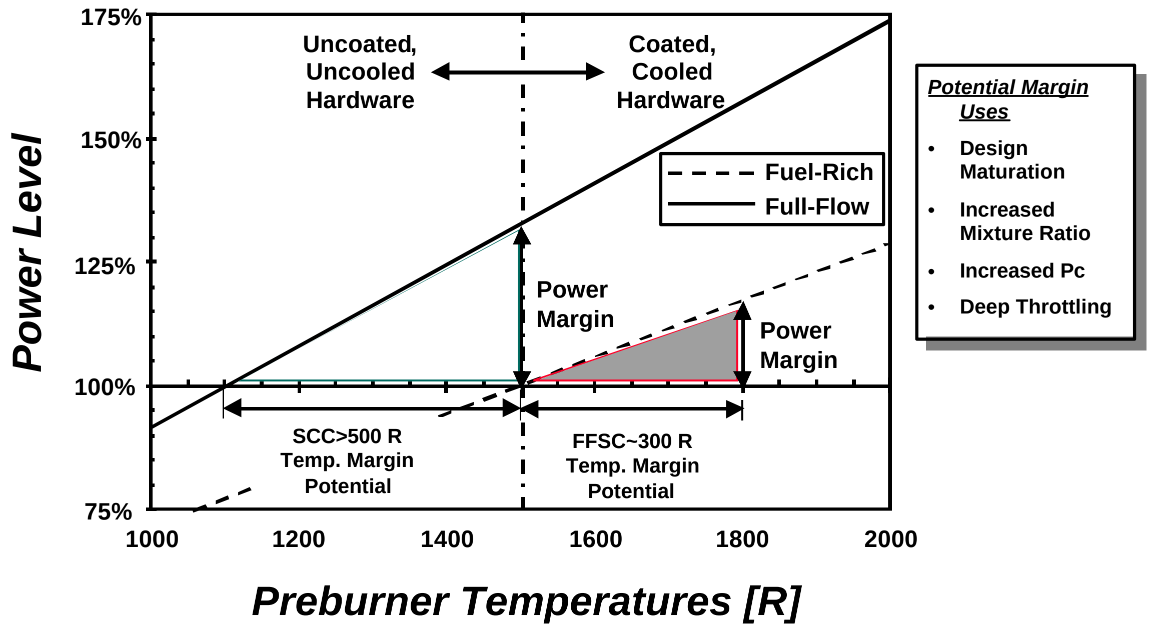

Figure 5. Power Margins Versus Cycles

These results show that significant engine margins can be designed into a future SSTO O_2/H_2 engine. These margins could be used to improve engine life, operability, and mission adaptability. The impact of the margins also provides valuable guidance for technology investment. For example, the incorporation of an oxidizer-rich preburner in the FFSC cycle is a key factor in reducing required turbine temperatures and improving engine life.

Engine Weight Calculation

The weight estimate was a bottoms up CAD design of the entire engine including all major and minor (e.g., drain lines, heat shield attachment flange) components. A design layout was generated for a FFSC cycle engine at 4,000 psi chamber pressure. The layout was then used for a more detailed weight determination as well as producing configuration drawings. A detailed weight statement of the SSME was used so that no element of a real, fielded, reusable engine was unaccounted for.

The weights included all the engine systems that would be in a reusable engine such as the SSME. Thus controllers, line insulation, gimbal attachments, drain lines, etc. were included. Installation specific systems such as the gimbal actuators and the engine heat shield were not included in the calculated engine weight. However, these items were explicitly calculated by the vehicle weight code.

The engine weight was calculated with a moderate number of near and midterm technologies included in the new engine. The new technology used was jet pumps as the boost pumps, turbomachinery specifically designed to lower cost and weight, EMA valves, and a limited use of advanced materials for the thrust cone, gimbal bearing, H_2 valve bodies, H_2 pump, gimbal actuator attach bracket, support struts, and the nozzle jacket.

Because few advanced materials were used and only for a few major engine components, there was weight margin in the estimate compared to methods which emphasize material approaches to lowering engine weight. Indeed, the material used for oxygen rich combustion gas compatibility, Haynes 214, is a relatively low strength material which will probably be replaced in future designs. This material was chosen to produce uncoated, long life operation even at the cost of additional weight.

Figure 6 shows the design point and characteristics of the engine. Figure 7 shows the procedures used for the weight calculation and Figure 8 shows the results.

- Design Point
  - Cycle – FFSCC
  - Chamber Pressure – 4,000 psi
  - Sea Level Thrust – 421,000 lbf
  - Area Ratio – 70.62
  - Fuel Turbine Operating Temperature – 1,100 ÞR
  - Oxidizer Turbine Operating Temperature – 1,100 ÞR

- Characteristics
  - Fuel Rich Fuel Turbopump
  - LOX Rich LOX Turbopump
  - Jet Pump Low Pressure Pumps
  - Propellant Duct Gimbal Accommodation on Vehicle Side
  - SLIC™ Turbomachinery
  - Uncooled Powerhead
  - EMA Valves
  - Preburner Injectors Gas/Liq Impinging Jet
  - MCC Injectors Gas/Gas Co-Ax
  - Redundant Laser Igniters
  - Autogenous Pressurization on Both Sides
  - Pump Conditioning Fluid Recirculated to Tank on Both Sides

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Figure 6. Design Point and Characteristics for Weight Calculations

- Overall Procedure
  - Various Individual Design Procedures Combined at CATIA Assembly Level for Packaging and in Spreadsheet for Weights

- Two Direct Design Procedures are Used

  - CATIA Solid Model (e.g., Hot Gas Manifold)
    - Designed as Individual Component
    - Wall Thickness Calculated
    - Minimums Applied in Model
      - 1.5 Factor for Dynamic Loads Applied to Wall Thickness if Appropriate
    - Solid Volume Returned to Spreadsheet for Weights
    - In Spreadsheet
      - Density used on Solid Volume for Weight
      - 1.02 Factor and 1.05 Factor Applied to Weight

  - CATIA Assembly Model (e.g., Duct)
    - Designed at Assembly Level for Dimensions, Clearances, and Packaging
    - Dimensions Returned to Spreadsheet for Weights
    - In Spreadsheet
      - Wall Thickness Calculated and Minimums Applied
      - Other Subcomponents Calculated (Flanges, Insulation, Insulation Shields, etc.)
      - Weights Calculated from Material Choices and Dimensions

- Other Procedures are Used For Some Components and May be Combined

  - Scaled (e.g., Valves)
  - Outside Reference (e.g., STME-100 for Controller)
  - Outside Model or Correlation (e.g., SLIC™ Turbomachinery)
  - Directly from SSME (e.g., Static Seals)

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Figure 7. Weight Calculation Procedure

Advanced Booster Engine 4k Pc O2/H2

Weight Breakdown

- **Vacuum Thrust** **484,585**
- **Sea Level Thrust** **421,000**
- **H2/O2 Core Pc = 4000** **Nozzle exp. ratio 70**

DUAL MIXED PRE-BURNERS

Main Combustion Chamber					**496**
with injector and liner			CR = 2.92		
			NARloy	NiCo	
Regenerative Cooled Nozzle					**625**
			A-286	Titanium	
Turbopumps					
HPFP	SLIC	**499**	n• AL | n• AL	Thermo-Span | RIM-D1, A286 TMP	
HPOP	SLIC	**562**	INCO 718 | INCO 718	Haynes 214 | Haynes 214	**1061**
Pre-Burners					
FPB		**40**		Thermo-Span	
OPB		**334**		Haynes 214	**374**
Valves					**361**
Propellant Ducts					
FUEL		**265**		INCO 903	
includes repress., pump recir., drain, & cryo purge					
OXID		**358**		INCO 718	**623**
includes repress., pump recir., drain, pogo systems, & O2 hxr					
Fuel Hot Gas Manifold				Thermo-Span	**262**
Ox Hot Gas Manifold				Haynes 214	**198**
Controller, Harness, Sensors, & Ignition					**150**
Structure					**252**
Bolts & Misc. parts					**165**
			TOTAL		**4,567 lbs**
					106.10 Tvac/W
					92.18 Tsl/W

Figure 8. Weight Results - Long Life, High T/W O_2/H_2 Rocket Engine

Engine Life Calculation

This engine is designed specifically to reduce maintenance and increase life over current designs. This is achieved primarily through less stressful internal environments, such as much lower turbine temperatures, replacement of low pressure turbomachinery with jet pumps, use of simplified turbomachinery without interpropellant seals, and integrated health monitoring systems.

The development of this power cycle also includes oxidizer rich operation and major turbomachinery thrusts which also will have significant impacts on rocket engine life and maintenance.

Turbomachinery technology for advanced rocket engines has concentrated on simplifying the machinery to reduce part counts, development time, production costs, weight, and improve operations and life.

The use of mixed preburner cycles will also reduce the turbine temperatures needed, which will very greatly improve turbopump life.

The designs implementing the above thrusts also strive to reduce part counts. New designs have ~30 to 50 major parts compared to ~200 to 250 for SSME turbopumps. The part count reduction reduces development time and production costs and improves operability.

There are also some new materials being used, such as nanophase Al, RIM-D1, Haynes 214, and Thermo Span, which improve properties or eliminate the need for coatings.

Lastly, those new engines using separate low pressure pumps (such as the SSME) will replace them with jet pumps. This will eliminate an entire set of complex turbopumps with the replacements being simple, no moving parts devices. The net effect will be a further improvement in operations and life and a further reduction in production costs.

The net effect of these changes is shown in Figure 9 which shows the predicted mean time between failures in flights for this engine.

Engine Reliability Impact

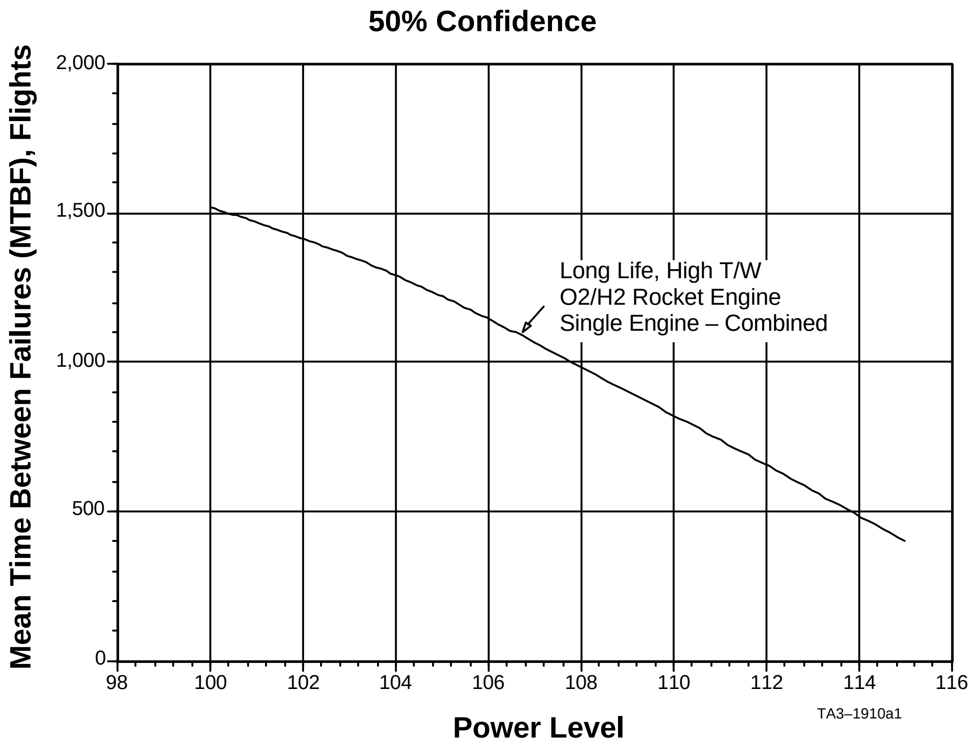


Figure 9. Engine Life Prediction

Spaceliner Architecture / System / Subsystem Application(s): This technology applies to all rocket engine main propulsion elements in any of the Spaceliner 100 architectures. It would also apply to the rocket elements of combined cycles, either as embedded elements or as separate (tail) rockets. Subsets of it would also apply to OMS and RCS propulsion.

Current Technology Readiness Level (TRL):

OVERALL PROPULSION SYSTEM TECHNICAL MATURITY

TRL = 5+ for basic component designs, 4 for nanophase Al

Maturity without use of new materials

Gas/Gas Injection	TRL	5+
Ox Rich Preburner	TRL	5+
H_2 Rich Preburner	TRL	6-9
Laser Ignitors	TRL	5
Combustion Chamber	TRL	6-9
Nozzle	TRL	6-9

Nozzle Radiation Skirt	TRL	7
EMAs	TRL	6
Sector/Showerhead Valves	TRL	6
Low Part Count, Hydrostatic Bearing, No Interpropellant Seal Turbopumps (SLIC®)		
Up to 3 Stages	TRL	5
4 Stages H_2	TRL	4
Predictive, Adaptive, Integrated Condition/Health Monitoring		
Parts	TRL	6
Fully Integrated Concept	TRL	3-

Maturity of new materials for rocket engine application

Nanophase Al	TRL	4

Summary

The system components without nanophase Al and health/ /condition monitoring has a TRL of mostly 5 and would produce an engine with higher thrust/weight than SSME or proposed RLV engines (92 sea level) and would have a long life (>200 flights) and much lower maintenance than the SSME.

The TRL is 4 for the nanophase Al in the H_2 pump but the risk is limited to the implementation of the materials, not the basic component designs. The cost of failure would be a slight weight increase in the H_2 turbopump.

The TRL for the Predictive, Adaptive, Integrated Condition/Health Monitoring is low and the low operating costs depend on it. However, the need for this system is the same for all Spaceliner 100 propulsion technologies and thus the TRL of the monitoring technology does not discriminate against this or any other propulsion technology for Spaceliner 100.

Aessment of Major Propulsion System Elements:

Element Name: Combustion Chamber

Description = Regen cooled chamber, gas/gas injection, H_2 coolant. Combustor extends to a TBD area ratio based on future heat transfer analysis.

Approximate Number of discrete "piece parts" comprising the Element = Parts count is ~3,800 compared to ~7,600 on a SSME.

Critical Technology Requirements = Gas/Gas injection

Current Element Technical Maturity (TRL Level) = 5+

Element Name: Nozzle

Description = Tubed regen nozzle.

Approximate Number of discrete "piece parts" comprising the Element = Parts count ~80% of current designs.

Critical Technology Requirements = None

Current Element Technical Maturity (TRL Level) = 7

Element Name: Preburners

Description = Uncooled. Temperatures ~1,200 R . Gas/liquid injectors.

Approximate Number of discrete "piece parts" comprising the Element = Parts count ~5 for each preburner.

Critical Technology Requirements = Ox rich preburners

Current Element Technical Maturity (TRL Level) = 5+

Element Name: O_2 Turbopump

Description = Inco 718 pump rotor, Haynes 214 turbine rotor, Haynes 214 turbopump housing, no coatings, 2 stage pump, hydostatic bearings, no interpropellant seal, liftoff seal.

Approximate Number of discrete "piece parts" comprising the Element = Parts count ~30 (much lower than current designs).

Critical Technology Requirements = Ox rich operation

Current Element Technical Maturity (TRL Level) = 5

Element Name: H_2 Turbopump

Description = Nanophase Al pump rotor, RIM-D1 turbine rotor, Al pump housing, Thermo-Span turbine housing, no coatings, 4 stage pump, hydrostatic bearings, no interpropellant seal, liftoff seal.

Approximate Number of discrete "piece parts" comprising the Element = Parts count ~50 (much lower than current designs)

Critical Technology Requirements = 4 stage pump design, nanophase Al parts and fabrication processes

Current Element Technical Maturity (TRL Level) = 4

Element Name: Valves

Description = Sector/showerhead valve with EMA actuators.

Approximate Number of discrete "piece parts" comprising the Element = Part count typical of current designs

Critical Technology Requirements = None

Current Element Technical Maturity (TRL Level) = 6

Element Name: Propellant Ducts

Description = H_2 propellant ducts fabricated from Thermo-Span, O_2 from Haynes 214.

Approximate Number of discrete "piece parts" comprising the Element = Part count similar to current designs

Critical Technology Requirements = None

Current Element Technical Maturity (TRL Level) = 6

Element Name: Hot Gas Manifolds

Description = H_2 rich hot gas manifolds fabricated from Thermo-Span, O_2 rich

from Haynes 214.
Approximate Number of discrete "piece parts" comprising the Element = Part count ~3 per manifold
Critical Technology Requirements = None
Current Element Technical Maturity (TRL Level) = 6

Investments Required to Mature the Technology for Spaceliner.

Provide a summary estimate of the dollar levels of investment by Government Fiscal Year (FY) beginning with FY 2001 to mature the candidate technology sufficiently for incorporation in a flight demonstrator *system* (at a TRL 6 for reference) which could include any and all elements of a Spaceliner 100 architecture.

The use of full flow staged combustion cycles in future advanced O_2/H_2 engines will allow less severe internal pressures, temperatures, and stresses; wider component trade space; and more component margins and thus lower maintenance and longer life due to the increase of power availability of using both oxidizer and fuel flows to power the turbomachinery. These are crucial needs for developing rocket engines capable of supporting the Spaceliner 100 cost goal.

Full flow staged combustion cycle technology is currently under development in the Air Force Integrated Powerhead Demonstration (IPD) and Integrated High Payoff Rocket Propulsion Technology Initiative (IHPRPT) programs.

The development of this power cycle also includes oxidizer rich operation and major turbomachinery thrusts which also will have significant impacts on rocket engine life and maintenance.

Turbomachinery technology for advanced rocket engines has concentrated on simplifying the machinery to reduce part counts, development time, production costs, weight, and improve operations and life.

New turbopump designs will use fluid film bearings which eliminates the bearing DN speed limit, which, in turn, allows smaller diameter turbopumps. They will make use of mixed preburner drive cycles (using fuel rich gases to power the fuel turbopump and oxygen rich gases to power the oxygen turbopump) which will allow the elimination of interpropellant seals which both shortens the turbopumps and eliminates a criticality 1 failure mode. These two thrusts also make the turbopump both shorter and smaller in diameter, which, because the weight is mostly in the turbopump housing, will greatly reduce future turbopump weights.

The use of mixed preburner cycles will also reduce the turbine temperatures needed, which will very greatly improve turbopump life.

The designs implementing the above thrusts also strive to reduce part counts. New designs have ~30 to 50 major parts compared to ~200 to 250 for SSME turbopumps. The part count reduction reduces development time and production costs and improves operability.

There are also some new materials being used, such as nanophase Al, RIM-D1, Haynes 214, and Thermo Span, which improve properties or eliminate the need for coatings.

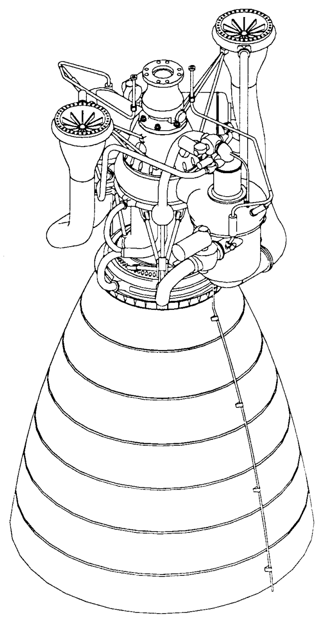
Lastly, those new engines using separate low pressure pumps (such as the SSME) will replace them with jet pumps. This will eliminate an entire set of complex turbopumps with the replacements being simple, no moving parts devices. The net effect will be a further improvement in operations and life and a further reduction in production costs.

Figure 10 shows an estimate of the costs required to demonstrate hot-fire operation in the IHPRPT Phase 1 Demo and the further costs to develop the Long Life, High T/W O_2/H_2 Rocket Engine. The development includes a first shipset of five engines.

Years	1	2	3	4	5	6	7	8	9	10
TRL 2: Basic principles observed and technology concept formulated										
TRL 4: Component and/or breadboard validation in laboratory environment		$10M								
TRL 6: Prototype demonstration in a relevant environment					$20M					
TRL 8: System flight qualified through test and demonstration								~ $650M		

Figure 10. Technology Implementation Plan - Long Life, High T/W O_2/H_2 Rocket Engine Using FFSC Cycle

Potential Benefits of the Technology to Spaceliner: A H_2/O_2 engine with SSME class performance and a thrust/weight (T/W) ~90 enables the SSTO mission using well developed rocket engine technology without requiring new and untried technologies such as combined cycles and/or pulsed detonation engines. Achieving such a T/W lowers the vehicle dry and gross takeoff weights enough that it is on a much less sensitive portion of the vehicle performance versus weight curve. Figure 11 describes the engine that was generated under the two NASA-MSFC contracts listed in "Information References" below.



• Sea Level Thrust, lbf	421,000
• Vacuum Thrust, lbf	484,585
• Nozzle Exit Pressure, psi	4.5
• Area Ratio	70
• Propellants	O_2/H_2
• Mixture Ratio, O/F	6.9
• Power Cycle	Full Flow Staged Combustion
• Turbine Temperatures	
• Ox Rich, ÞR	1,100
• Fuel Rich, ÞR	1,150
• Chamber Pressure, psi	4,000
• Seal Level Specific Impulse, sec	392.2
• Vacuum Specific Impulse, sec @ 6.9 MR	451.4
@ 6.0 MR	456.4
• Sea Level Engine Thrust/Weight	92
• Regen Cooled Bell Nozzle	

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Figure 11. Baseline Long Life, High T/W O_2/H_2 Rocket Engine

The performance of this engine in the SSTO mission using a cylindrical winged body vehicle in a vertical takeoff/horizontal landing mode is described in Figure 12.

- **NOTIONAL VEHICLE SYSTEM SUMMARY - 40,000 lb payload**

Vehicle Description	Winged Cylindrical Body (CONSIZ-WB000) Vertical Takeoff/ Horizontial Landing Single Stage-to-Orbit with no Strap-ons or Expendables
Orbit	100 nmi, circular, 28.6Þ
Ideal ΔV Needed (Includes Losses for Gravity, Drag, Steering, etc.)	29,307 ft/sec
Mission Average Isp	439.6 sec
Payload	40,000 lb
Payload Bay	15 ft x 30 ft
Vehicle Sea Level Thrust/Weight	1.2
GLOW	1,977,650 lb
Dry Weight	182,218 lb
Dry Weight Margin	78,714 lb
Usable Margin (Dry Weight Equal Reference Vehicle Dry Weight)	47,277 lb
GLOW at Maximum Usable Margin	3,079,341 lb
Length	157.3 ft
Body Width	31.4 ft
Wing Span	87.0 ft
TanksAl-Li	
Lox Tank	Forward*
RCS/OMS/Main Propulsion	O2/H2 Common Tanks**
Engine Mixture Ratio, O/F	6.9
Engines	4 @ 593,295 lbf Sea Level Thrust (Lower Cost) 7 @ 339,026 lbf Sea Level Thrust (Engine Out Capability) (Scaled Versions of Engines in Table 1)
Development Weight Margin	15%
Fielded Weight Margin	0%
Reference Baseline***	Bipropellant, Access to Space
GLOW	2,719,864 lb
Dry Weight	260,930 lb

* Sized as forward; would be better operationally if Lox tank were aft.
** CONSIZ not modified to reflect the weight impact of common tanks.
*** Access to Space engine run to HRST orbit and payload using version of CONSIZ used for this study (versus Access to Space results of 2,449,497 lb/233,292 lb).

Figure 12. Nominal Vehicle Performance with Long Life, High T/W Rocket Engine

A comparison of the performance of this engine against other propulsion concepts being considered for Spaceliner 100 was made. Figures 13 and 14 show the results.

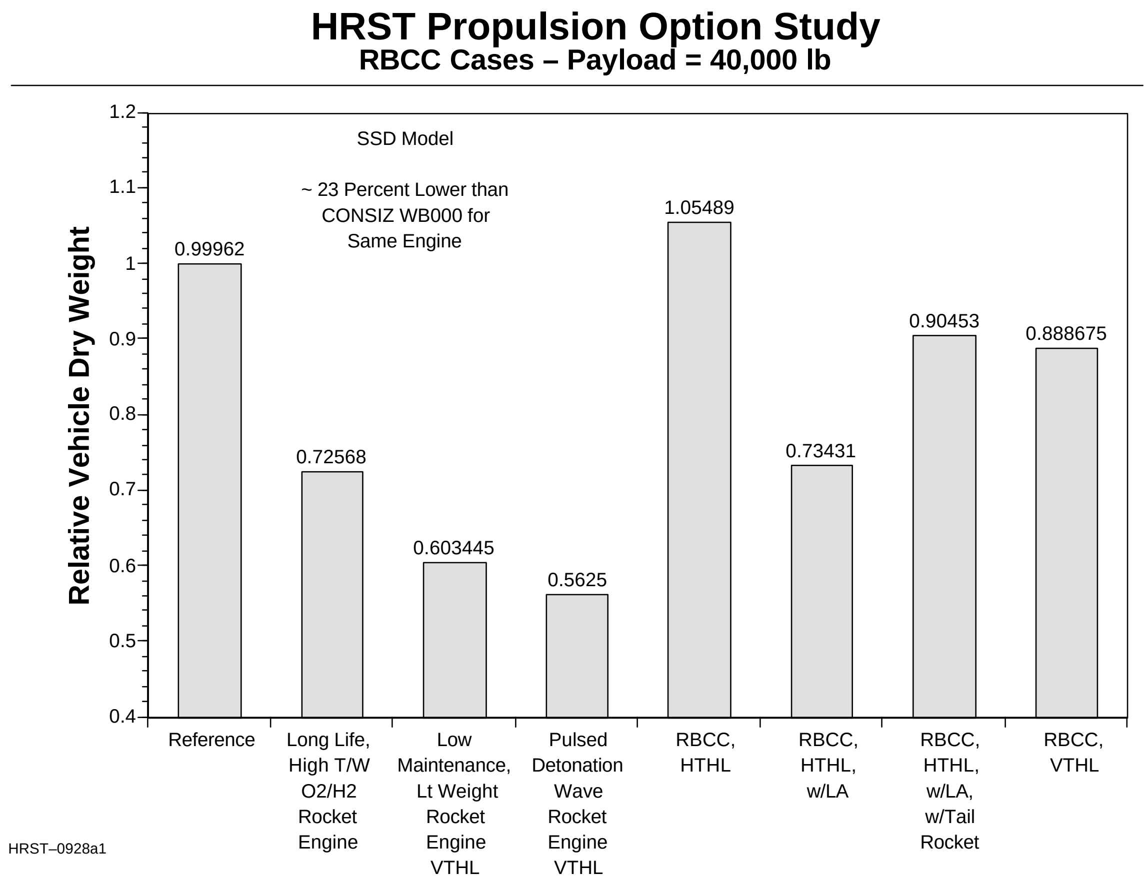

Figure 13. Comparison of Long Life, High T/W O_2/H_2 Rocket Engine to Other Spaceliner 100 Propulsion Concepts - Vehicle Dry Weight

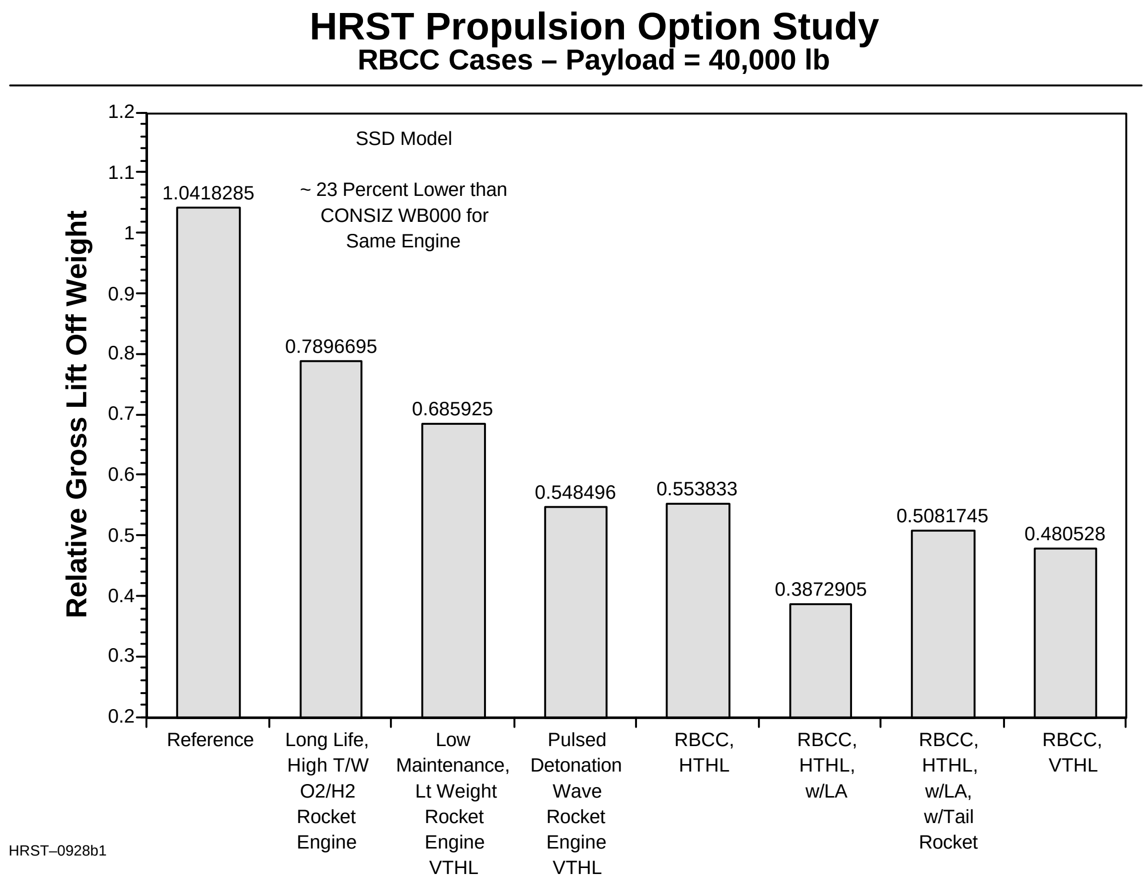

Figure 14. Comparison of Long Life, High T/W O_2/H_2 Rocket Engine to Other Spaceliner 100 Propulsion Concepts - Gross Lift Off Weight

The reference in these two charts is the Access to Space rocket vehicle using a variation of the SSME. From these two charts, the Long Life, High T/W O_2/H_2 Rocket Engine produces dry vehicle weights comparable to, and in some cases, lower than RBCC engines for SSTO applications (the RBCCs used a very good engine thrust/weight of 32.7). On the other hand, the gross lift off weights are ~50% higher. Overall the Long Life, High T/W O_2/H_2 Rocket Engine is a viable candidate for the Spaceliner 100 application and it has far less risk than any other propulsion approach.

Potential Risks in Developing the Technology: There are minimal risks associated with developing this technology.

The technology drivers for this technology are:

Development of the FFSC engine cycle to help produce long life and high margins;

LOX rich preburner and turbine operation;

Gas/gas combustion;

Development of low part count, hydrostatic bearing, no coating, no interpropellant seal turbopumps to produce long life, low maintenance, and low weight;
Use of nanophase Al for H_2 pump to produce low weight;
Development of four stage H_2 pump to achieve 4,000 psi chamber pressure.

The first two of these are being addressed by the Air Force and are well on their way to being developed. The third (nanophase Al) is also being developed by the Air Force. If it fails to meet all its requirements, then the effect would simply be a slight turbopump weight increase. The last item is not being addressed. There are alternate ways to address a four stage pump requirement including using two two stage pumps, new materials to get by with a three stage pump, or reduce the chamber pressure slightly, or some combination of these approaches.

The net effect is that this is a low risk technology approach to Spaceliner 100 which appears capable of meeting the Spaceliner 100 goals.

Information References: This engine concept has been developed, defined, and its applications studied under two NASA-MSFC contracts: NAS8-39210, "Advanced Transportation Systems, Alternate Propulsion Subsystem Concepts", and NCC8-113, "Highly Reusable Space Transportation Propulsion Option Study".

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