NASA-TM-111897

AIAA

AIAA 94-3317
Advanced Low-Cost O_2/H_2 Engines for the SSTO Application

B. David Goracke and Daniel J. H. Levack
Rockwell International Corporation
Rocketdyne Division
Canoga Park, CA

Robert F. Nixon
NASA Marshall Space Flight Center
Huntsville, AL

30th AIAA/ASME/SAE/ASEE
Joint Propulsion Conference
June 27-29, 1994/Indianapolis, IN

Advanced Low-Cost O_2/H_2 Engines for the SSTO Application

B. David Goracke* and Daniel J.H. Levack**
Rocketdyne Division, Rockwell International
Canoga Park, California

Robert F. Nixon†
NASA, Marshall Space Flight Center
Huntsville, Alabama

Abstract

The recent NASA Access to Space Study examined future ETO transportation needs and fleets out to 2030. The baseline in the Option 3 assessment was a single-stage-to-orbit (SSTO) vehicle. A study was conducted to assess the use of new advanced low-cost O_2/H_2 engines for this SSTO application. The study defined baseline configurations and groundrules and defined six engine cycles to explore engine performance. The cycles included an open cycle, and a series of closed cycles with varying abilities to extract energy from the propellants to power the turbomachinery. The cycles thus varied in the maximum chamber pressure they could reach and in their weights at any given chamber pressure. The weight of each cycle was calculated for two technology levels versus chamber pressure up to the power limit of the cycle. The performance in the SSTO mission was then modeled using the resulting engine weights and specific impulse performance using the Access to Space Option 3 vehicle. The results showed that new O_2/H_2 engines are viable and competitive candidates for the SSTO application using chamber pressures of 4,000 psi.

Nomenclature

EMA	electromechanical actuator
ETO	earth-to-orbit
F	fuel
FFSCC	full flow staged combustion cycle
GG	gas generator
I_{sp}	specific impulse
LOX	liquid oxygen
MCC	main combustion chamber
MFV	main fuel valve
MOV	main oxidizer valve
MR	mixture ratio
O	oxidizer
PB	preburner
PBFV	preburner fuel valve
PBOV	preburner oxidizer valve
P_c	chamber pressure
SCC	staged combustion cycle
SL	sea level
SSME	Space Shuttle Main Engine
SSTO	Single Stage to Orbit
vac	vacuum

* Director, Systems Architecture
** Program Manager, Advanced Systems Programs
† Manager, Upper Stages, Program Development

Background

Recent NASA transportation studies have examined a wide spectrum of liquid rocket engines to address either new or evolved capabilities, cost-effectiveness, or enhanced operational efficiency.[1,2] Most recently the NASA Access To Space study has assessed the future of NASA's ETO transportation needs and fleet out to 2030. The baseline considered in the Option 3 assessment, Advanced Technology, was a tripropellant $LOX/H_2/RP$ engine or a comparable LOX/H_2 engine.

A recent activity has been conducted to evaluate a new advanced low-cost LOX/H_2 engine for the SSTO application as part of the Advanced Transportation System Study Technical Area 3 - Alternate Propulsion Subsystem Concepts (NAS8-39210) contract sponsored by NASA. The purpose of this study was to provide an assessment of alternate/next-generation propulsion systems for

ETO and in-space vehicles and define the technology/advanced development needed. In particular, the study was to define next generation LOX/H_2 engines and evaluate their use for the SSTO mission.

The study, from Marshall Space Flight Center, defined engine concepts using O_2/H_2 with the specific objectives: to produce engine concepts which had high performance in order to reduce resulting vehicle life cycle costs through decreasing vehicle empty weights and to also lower engine life cycle costs through the component and operating parametric choices, the inclusion of specific technologies, and improved operations. An additional objective was to identify the key technologies needed to implement these engine concepts. This paper presents the results of the engine concept investigation and the resulting vehicle performance using the NASA Access-to-Space Option 3 vehicle as defined by NASA-LaRC.

Concept Definition

The study defined six engine cycles for study. Baseline component parameters (such as turbine material and operating temperature) were then chosen along with sets of variations on these parameters. A single position bell nozzle was chosen as representative (i.e., a different choice would not have produced differentiation among the cycles and concepts although it would have changed the overall vehicle performance). A fixed exit pressure of 4 psi was chosen since previous studies had shown that value to produce near optimum vehicle performance for a single position bell nozzle[3]. Consequently, engine area ratio was a set function of chamber pressure as shown in Figure 1. Figure 2 shows the baseline component parameters, their ranges examined, and other specific technologies included in the baseline. The component range studies which have so far been finished and are presented in this paper are marked in the figure.

The six engine cycles examined included one representative open cycle for comparison (a gas generator cycle) and five closed cycles. The open cycle (Figure 3) had the lowest engine weight, but also had a significant performance penalty in comparison to the closed cycles.

In a closed cycle the amount of energy which can be extracted to pump the propellants, and thus increase chamber pressure and engine specific impulse, 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 five closed cycles explored this range of energy extraction capability.

The first cycle (Figure 4) was a full flow mixed preburner cycle using individual preburners to power the fuel pump and the LOX pump. The fuel preburner was fuel rich and the LOX preburner was LOX rich. Thus potentially all of both the fuel and LOX flows were available. This cycle could extract the most energy for pumping and thus was capable of the highest chamber pressure. Because it had the most and the largest powerhead components it was also the heaviest cycle of the five at a given chamber pressure and nozzle area ratio.

The next cycle, both in the ability to extract energy and in weight (i.e., second heaviest), was a cycle which used all of only one flow (in this case, H_2) but also used preburners for both the fuel and LOX pumps. This staged combustion cycle (SSC) was very similar to that used for the Space Shuttle Main Engine (SSME). Figure 5 shows this cycle.

The third closed cycle was one which also used all of one flow (H_2), but only one preburner. The preburner was used to power the fuel pump because the fuel pump needs more horsepower than the LOX pump, and, consequently the cycle could extract more energy if the one preburner was used on the fuel side. A fuel expander (fuel using only the energy from regenerative cooling) was used to power the LOX pump. This cycle, the hybrid cycle, is shown in Figure 6.

The inverse of the hybrid cycle was also examined: a preburner powering the LOX pump and a fuel expander powering the fuel pump. This cycle is illustrated in Figure 7.

The last closed cycle examined was one using fuel expanders to power both the LOX and fuel pumps. This cycle had the least ability to extract energy and thus had a lower maximum chamber pressure. However, it also had the lightest engine weight of the closed cycles at a given chamber pressure and area ratio. The expander cycle is shown in Figure 8.

The defined cycles were examined from a chamber pressure of 1000 psi to the limit the cycle could produce by using the Rocketdyne balance code. At each chamber pressure the pump and turbine stages were varied and both pump discharge pressures and engine weight were minimized.

Engine Weight Calculation

Engine weights were calculated for all six cycles as a function of chamber pressure. They will later be calculated as a function of different turbine operating temperatures and materials. 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. Figure 9 shows the methodology used to calculate the engine weights. They were calculated for two levels of technology: one with minimal advancement over that used in the SSME (referred to as the "bracketing" weight set since it should be an upper bound on a new engine), and one with a moderate number of near and midterm technologies included in the new engine (referred to as the "aggressive" weight set).

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. Advanced materials were used for few major engine components and thus there is probably weight margin in the estimate compared to methods which emphasize material approaches to lowering engine weight.

Figures 10 and 11 show the engine weights for both sets of technology assumptions. The weights shown are for the highest turbine temperatures (which use Si_3N_4 turbine wheels and represent a new technology). Work in progress suggests that turbine temperatures in the 1,000–1,700 °R range will work as well, having essentially the same vehicle dry weight, and having engine weight increases of only 2-3 percent.

The three cycles with preburners on the fuel side, where the majority of the horsepower is needed, attained high chamber pressures: ~6,000 psi for the hybrid cycle, ~7,600 psi for the staged combustion cycle, and ~8,700 psi for the mixed preburner full flow cycle. This was as expected since these cycles were chosen specifically to explore extracting more energy as additional preburners and then both flows were used to power the turbomachinery. The inverse hybrid cycle, which did not have a preburner on the fuel side, was power limited at ~2,400 psi; and the expander cycle reached ~2,000 psi.

The engine weights minimized in the 2,500 – 3,500 psi range.

SSTO Performance

Twenty sets of resulting engine characteristics (weight, thrust, specific impulse, and mixture ratio) were sent to NASA LaRC to determine the vehicle gross and empty weights for a 25K payload to 220 n.mi. at 51.6° inclination and with a 15% weight margin on the engine weight. These parameters are consistent with the Access to Space Study. A non-linear regression analysis was performed on these results and the resulting equation used to predict the other engine cases.

Figure 12 shows the vehicle results for the six cycles.

The figure shows that vehicle performance was strongly dependent on chamber pressure below 4,000 psi. However, above this pressure, the increase in specific impulse barely offset the increase in engine weight and the vehicle performance was fairly flat.

From the figure it is seen that the expander and inverse hybrid cycles could not reach competitive chamber pressures and that the gas generator cycle, despite having the lowest engine weight, had enough specific impulse penalty that it was also not competitive. However, the other three closed cycles were all similar in vehicle performance because the engine weight differences among them was not large enough to significantly impact vehicle performance.

These advanced low-cost O_2/H_2 engines are also seen from Figure 12 to be competitive with the RD-704 as reported in the Access to Space Study report of January 1994.

Summary

This study has shown that new O_2/H_2 engines are feasible and competitive in the SSTO application and that chamber pressures above 4,000 psi are not required. Further work is being conducted to verify the weight estimates, to examine lower turbine operating temperatures, and to study the margin capabilities of the cycles. This work will be reported in future papers.

Acknowledgement

The authors would like to recognize the key contributions made by the following individuals to the detailed evaluations contained in this paper: Chris Erickson and Pei-Shiun Chen of Rocketdyne and Doug Stanley and Roger Lepsch Jr. of NASA-LaRC.

References

1. Stafford, Thomas P., et al., "America at the Threshold," Report of the White House Synthesis Group on America's Space Exploration Initiative, May 3, 1991, Washington, D.C.

2. "Access To Space Study Summary Report," January, 1994.

3. Wheeler, D. B., "High Performance O_2/H_2 Engine Definition Final Report," 18 October 1990, AI-TR-90-051.

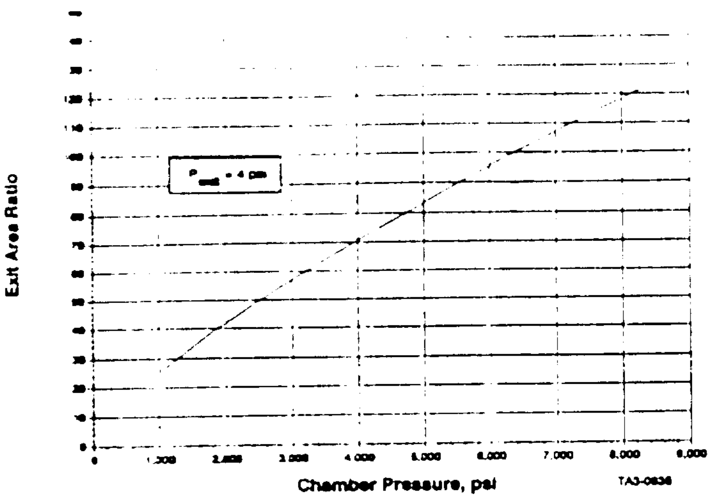

Figure 1. Exit Area Ratio Versus Chamber Pressure

- All Engines Calculated for 421,000 lbf Sea Level Thrust
- Combustion Efficiency
  - 0.995 at MR = 6*
  - 0.985 at MR = 7*, 10*
- Turbomachinery
  - Turbine Operating Temperatures
    - Fuel
      - 2,500 °R* — Si_3N_4
      - 1,000 – 1,900 °R — Astroloy
      - 1,000 °R — Al
    - Oxidizer
      - 2,500 °R* — Si_3N_4
      - 1,100 – 1,900 °R — Inco
  - Pumps
    - Use Boost Pumps
    - Use Kick Pumps Where Applicable
    - Fuel
      - 1 – 6 Stages (Al*, Ti)
    - Oxidizer
      - 1 – 4 Stages (Inco*)
- Fixed Bell Nozzle
  - Exit Area Ratio Sized for P_{exit} = 4 psi
- All Regenerative Cooling with H_2
- No Throttling Requirement

* Shown in this paper

Figure 2. Baseline Configuration Parameters

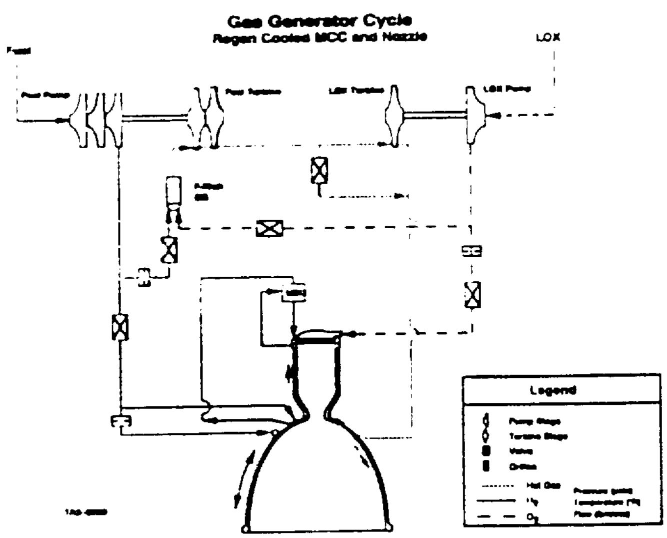

Figure 3. Gas Generator Cycle

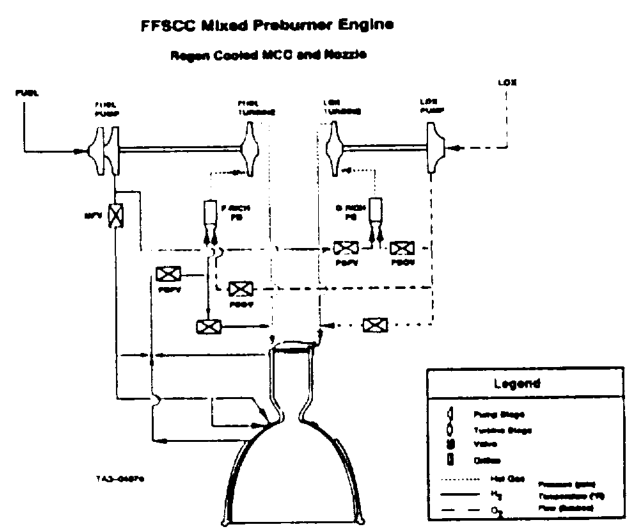

Figure 4. Mixed Preburners Full Staged Combustion Cycle (FFSCC)

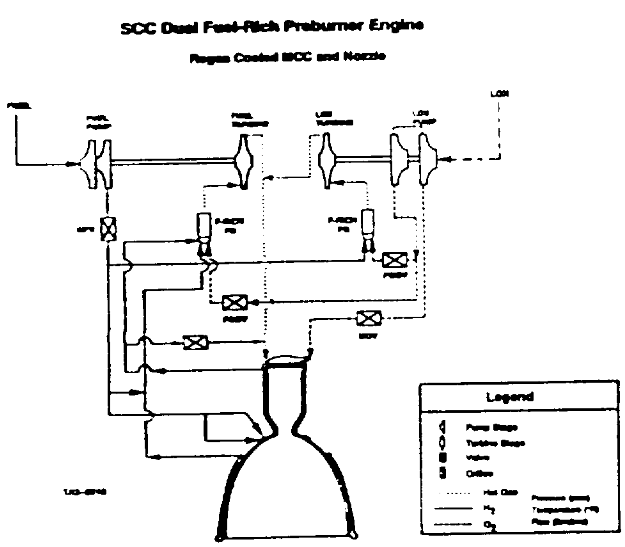

Figure 5. Staged Combustion Cycle (SCC)

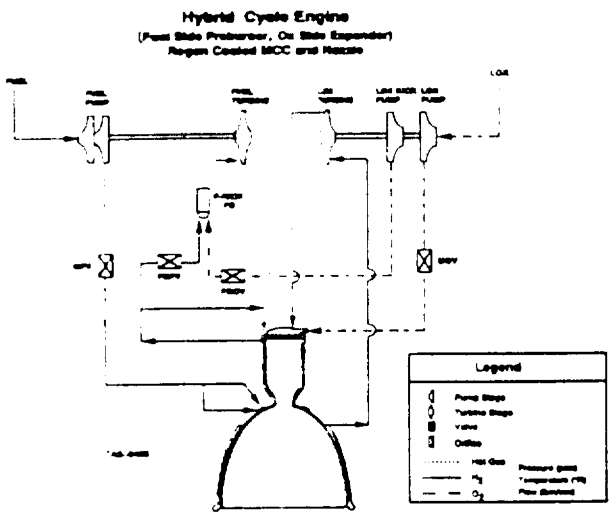

Figure 6. Hybrid Cycle

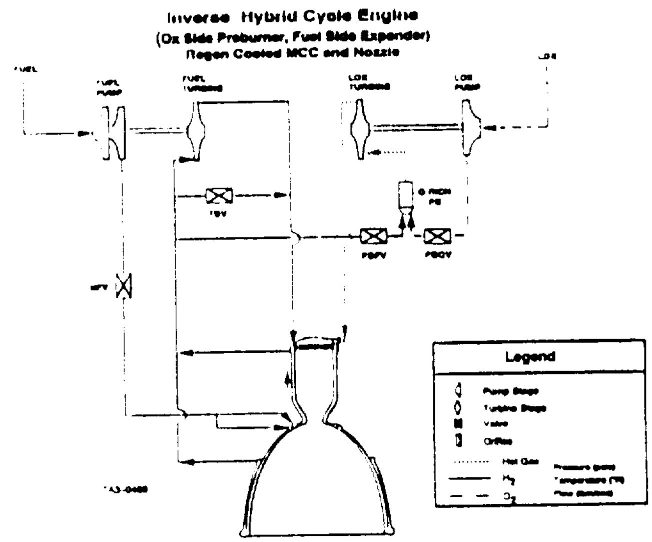

Figure 7. Inverse Hybrid Cycle

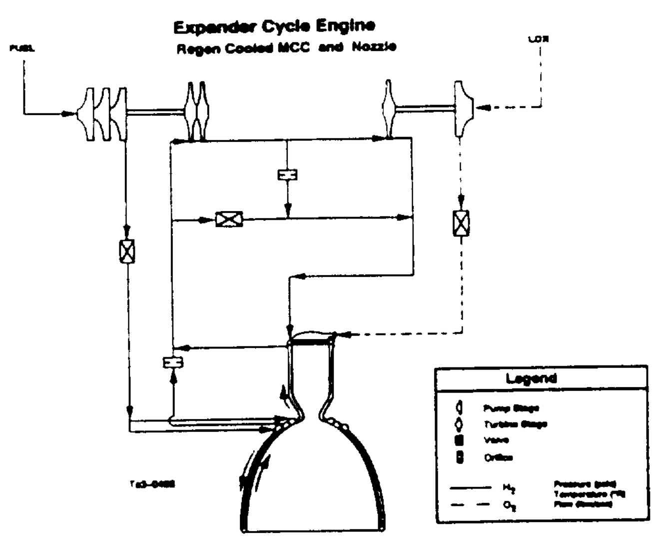

Figure 8. Expander Cycle

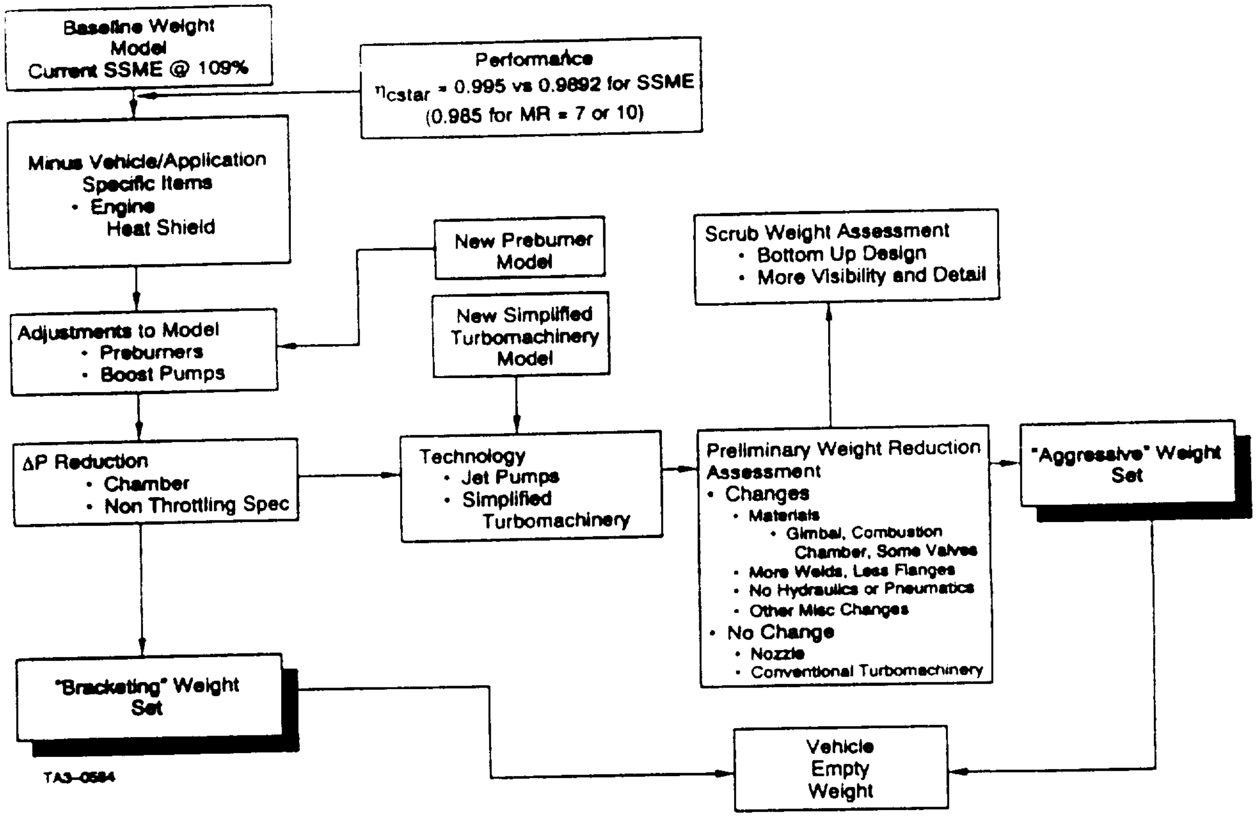

Figure 9. Engine Weight Generation Methodology

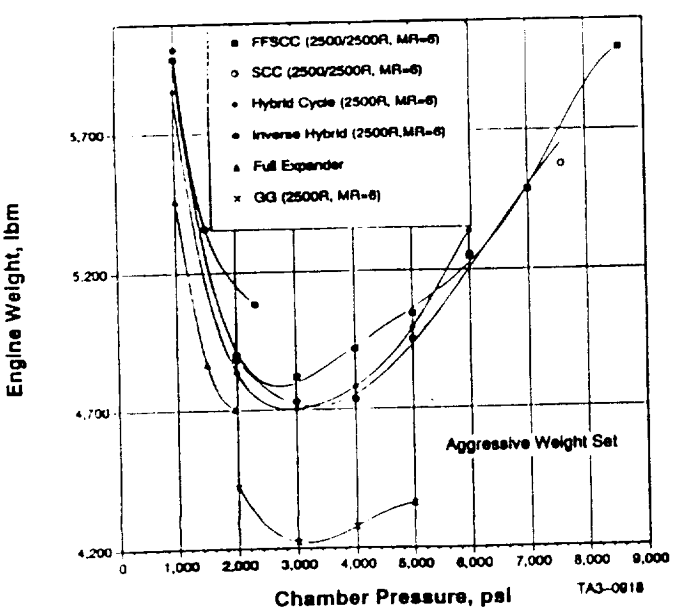

Figure 10. Engine Weights (Aggressive Set)

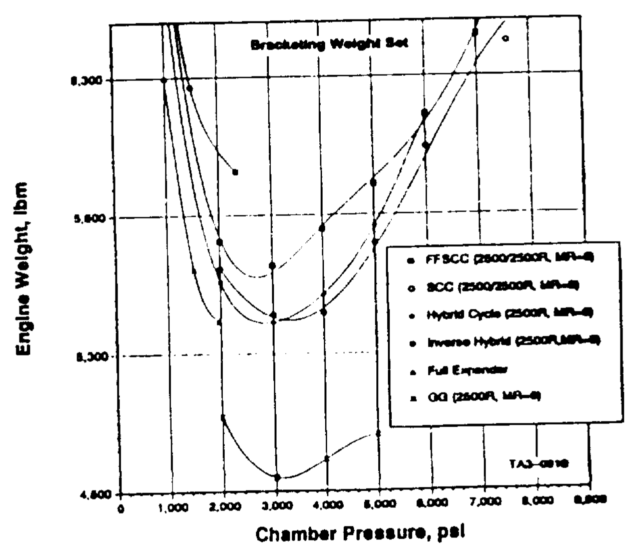

Figure 11. Engine Weights (Bracketing Set)

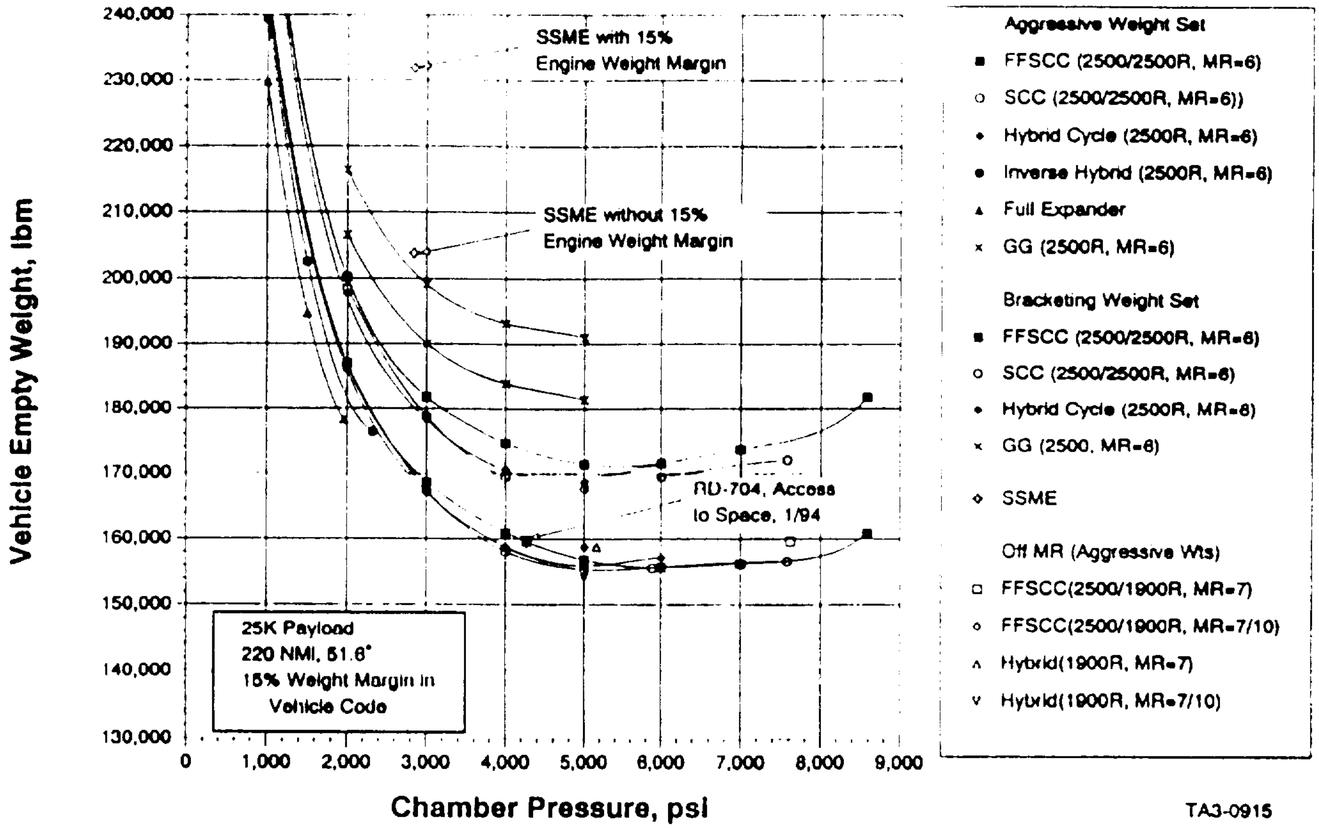

Figure 12. SSTO Performance of Advanced O2/H2 Engines