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Summary of Booster Propulsion/Vehicle Impact Study Results

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Authors:

Vincent A. Weldon *
Lawrence E. Fink **
Dwight U. Phillips **

ABSTRACT

Hydrogen, RP-1, propane, and methane have been identified by propulsion technology studies as the most probable fuel candidates for the boost phase of future launch vehicles. The objective of this study was to determine the effects of booster engines using these fuels and coolant variations on representative future launch vehicles. An automated procedure for integrated launch vehicle, engine sizing and design optimization was used to optimize two stage and single stage concepts for minimum dry weight. The two stage vehicles were unmanned and used a flyback booster and partially reusable orbiter. The single stage designs were fully reusable, manned flyback vehicles. Comparisons of these vehicle designs, showing the effects of using different fuels, as well as sensitivity and trending data, are presented. In addition, the automated design technique utilized for the study is described.

* Manager, Space Systems Preliminary Design, Boeing Aerospace and Booster Propulsion Vehicle Impact Study Manager
** Propulsion Flight Technology Engineer, Boeing Aerospace

INTRODUCTION

In order to compare the effects of different fuels and engines on a specific space launch vehicle concept, an approach was adopted in which alternative optimized configurations were developed to meet the same mission requirements. These optimized configurations were developed by simultaneous adjustment of the vehicle's engine and airframe variables to the demands of each other as well as to the performance requirements of the mission. Subsequently, the optimized configurations were compared to each other to determine the relative advantages and disadvantages of using different engine fuels on the vehicle concept.

To accomplish the optimization tasks economically, it is necessary to avoid the large number of design iterations required to analyze the effects of variable interactions using traditional parametric analyses (involving plots representing the effect of several variables on another). Boeing, therefore, developed a specialized analysis program called HAVCD (Hypervelocity Aerospace Vehicle Conceptual Design) to accomplish the study. This program combines launch vehicle design subprograms with a modified version of a previously developed optimization technique (reference 1) to perform the optimization analysis with only a small fraction of the number of design evaluations required by traditional parametric comparison methods.

Two-Stage Vehicles

The configuration concept selected for the two-stage vehicle comparisons uses a winged, flyback booster and an unmanned partially reusable "orbiter" core stage. The reference payload is 150,000 lb to a Space Station located in a 220-nmi circular orbit at an inclination of 28.5 degrees. The payload bay envelope is 33 ft in diameter by 70 ft long, permitting two Space Shuttle or two Titan IV payloads to be installed side-by-side. Figure 1 depicts a typical unmanned, partially reusable two-stage vehicle configuration of the type used for the two-stage concept comparisons.

Typical mission operations are also summarized on Figure 1. Lifting off vertically, the vehicle accelerates to a staging velocity, for minimum dry weight, of about Mach 5 with the booster and orbiter elements firing in parallel, but without propellant crossfeed. After staging, the winged booster flies back to a runway near the launch site using onboard automatic flight guidance and control and powered by flyback turbofan engines fueled by JP. The orbiter element continues its acceleration to orbit under the power of four Space Shuttle main engines (SSME's). At a dynamic pressure of 5 lb/ft^2 the payload shroud is jettisoned. The propulsion/avionics (P/A) module (Figure 1) houses the SSME's

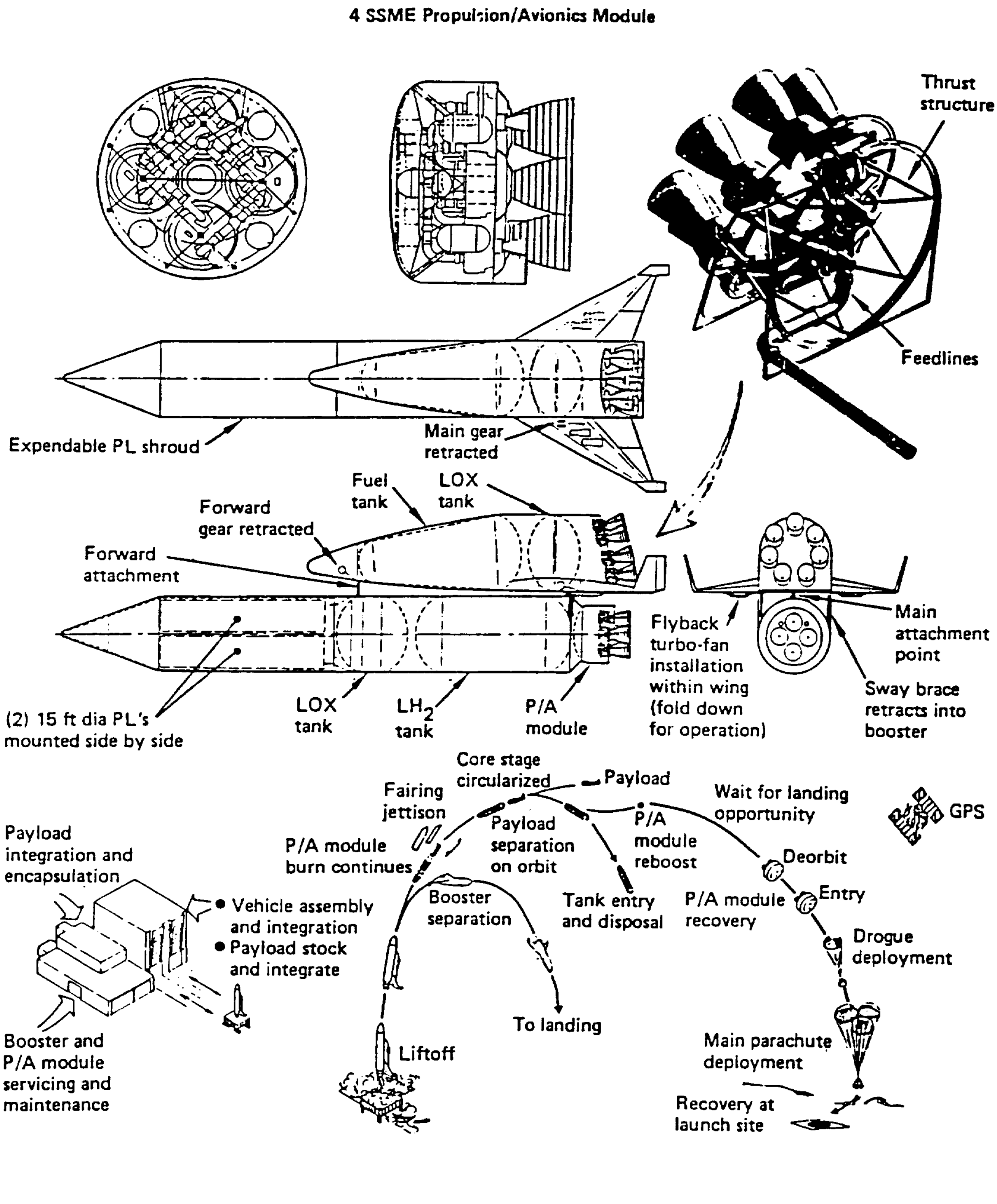


Figure 1. Two-Stage Partially Reusable Launch Vehicle Typical Features

and the Orbital Maneuvering System (OMS). The OMS uses Earth storable propellants and two engines located at the base of the orbiter stage for all maneuvers and deorbit burns. After payload placement in the proper orbit, the second stage propellant tanks are deorbited into their designated impact area, and the P/A module is deorbited into its recovery area. The P/A module, having a low L/D, thermally protected shape, reenters intact. After aerodynamic deceleration of the P/A module, parachutes are deployed to facilitate recovery near the launch site. The P/A module, as well as the booster, are later refurbished for reuse on a later flight.

The propellants, as well as the number and size of main engines, were varied for the flyback booster for each of the study configurations. The aft LOX tank of these boosters is cylindrical with elliptical domes. The fuel tank of the boosters is located forward of the LOX tank and is of a tapered cylinder shape to allow sufficient aerodynamic efficiency, for a given booster length, such that a forward canard is not required. In cases where an engine coolant is used that is different from the fuel, a third, smaller tank is located forward of the fuel tank in the booster. The booster fuselage fineness ratio (or length/diameter) is the same for all configurations; a value of about 4.5 was selected for adequate aerodynamic performance while permitting a low wetted area.

The fold-down large diameter turbofan engines of the booster are located within the wing and require a slightly protruding fairing (located on the underside of the wing). All subsystems, including structures, hydraulics, pneumatics, avionics, electrical, etc.) were assumed to reflect weights consistent with 1990 technology availability.

The tankage section of the orbiter element is of conventional design with the LOX tank being cylindrical with elliptical domes. The LH2 tank is a tapered cylinder with elliptical domes. The attachment structure to the booster is minimized on this element to increase its propellant mass fraction (since the orbiter element provides the majority of the ascent delta-velocity).

Single-Stage-to-Orbit Vehicles

The selected configuration design for a rocket powered, manned single-stage-to-orbit system is a fully reusable vertical takeoff, horizontal landing concept. A reference mission of 10,000 payload delivery to a 100-nmi circular polar orbit from WTR launch was also selected and the payload bay was sized to accommodate a 15-ft diameter by 30-ft long payload. A crew size of two was also assumed.

A typical operations profile and the selected configuration arrangement are shown for the single-stage-to-orbit vehicle on Figure 2. After liftoff and insertion into the proper orbit, the payload is deployed. Upon completion of the orbital mission the vehicle is deorbited and glides (unpowered) to a runway landing near the launch site for refurbishment prior to a later flight.

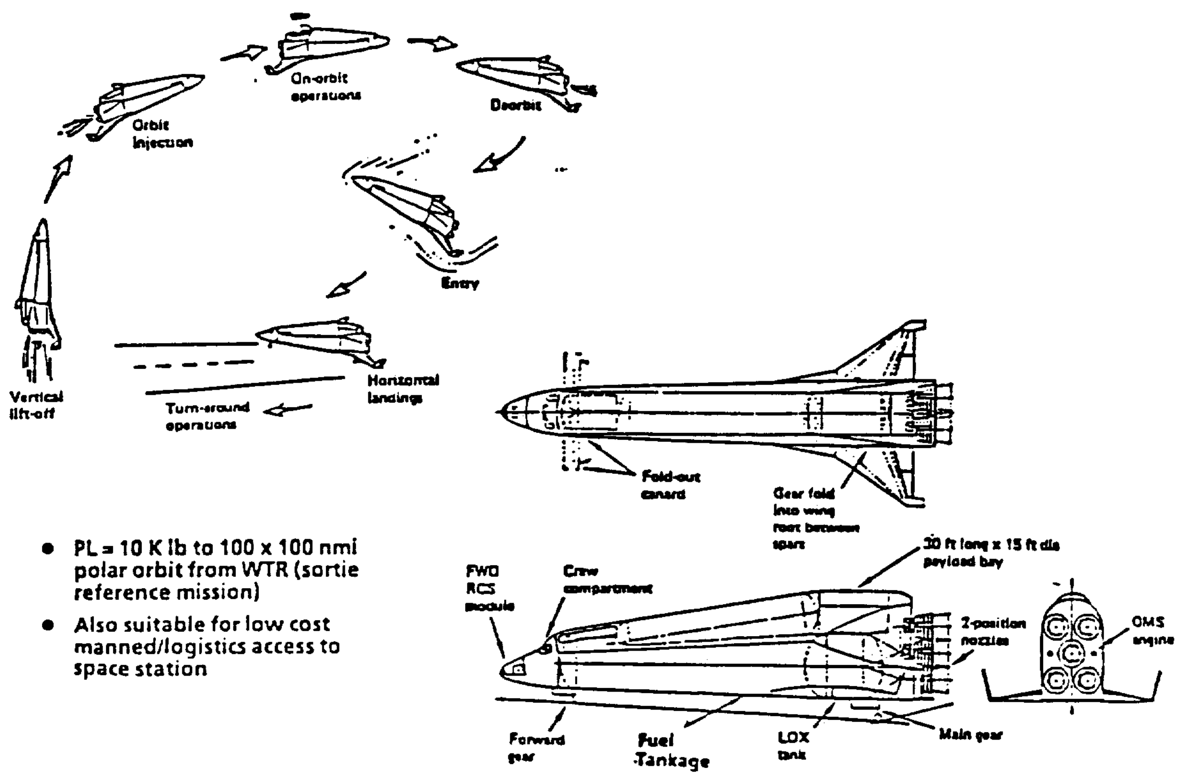


Figure 2. Reusable Single-Stage-To-Orbit Vehicle Typical Features and Operations Profile.

The single-stage-to-orbit vehicle has a forward, tapered fuel tank and an aft LOX tank (Figure 2). The area forward of the fuel tank houses the crew compartment, a deployable canard (for low-speed stability and control), and the nose landing gear. The payload bay is located above the LOX tank and near the vehicle center of gravity. The aft fuselage of the vehicle contains the thrust structure and engine feedlines. Most subsystems are relatively advanced and assume technology availability around the year 2000 or later.

DESIGN COMPUTER PROGRAM

The HAVCD (Hypervelocity Aerospace Vehicle Conceptual Design) computer program, mentioned earlier, was used to conduct design optimizations and generate trade data for this study. This program was developed to analyze the configurations of interest in this study.

HAVCD uses six specialized conceptual/preliminary design type subprograms as follows:

a. AIREZ - aerodynamics.
b. PROP - engine geometry, weights, and performance.
c. TAVB - airframe and subsystem weights.
d. ELES - tankage sizing and pressurization system.
e. NTOP - trajectory performance.
f. FLYBACK - flyback system design.

AIREZ relies on a blend of simplified aerodynamic theory and empirical relationships which result in acceptable agreement with wind tunnel test data. The subprogram generates a table of axial and normal aerodynamic force coefficients as a function of Mach number (Mach 0.3 to 20) and angle of attack (-10 to 60 degrees) based on airframe geometry determined by TAVB.

PROP was modified for this study to use the engine models from:

a. UTC/P&W, "Hydrocarbon Rocket Engine Study," contract NAS8-36355.
b. Rocketdyne, "Hydrocarbon Engine Study,." contract NAS8-36357.
c. Aerojet, "Hydrocarbon Engine Study," contract NAS8-36359.
d. Aerojet, "STME Configuration Study,." NAS8-3867

Besides computing engine specific impulse, nozzle and engine geometry and weight, it also computes the fuel/coolant/oxidizer split for the tanks of the vehicles based on the output of the trajectory subprogram.

TAVB was previously developed under IR&D by the Boeing Military Airplane Company for analysis of a specific type of vehicle. For purposes of this study, the same basic equations were modified to accommodate both the single-stage and two-stage vehicles described above. Conceptual design equations for the expendable tankage used in the two stage vehicle were provided by the Boeing Aerospace Weights Analysis technical staff.

ELES (Extended Liquid Engine Simulation) was written by Aerojet under Air Force contract (reference 3). Only the tankage, feedline, and pressurization system sizing and weight models were used in this study since preference was given to the modeling of other items in TAVB.

NTOP (New Trajectory Optimization Program) was the trajectory program used in this analysis. The trajectory is integrated using a point mass model. A perigee altitude of 50 nmi. was chosen to be low enough for good trajectory performance yet not be so low as to introduce unaccountable aerodynamic drag errors in the orbit circularization calculations. Propellant requirements for an orbit circularization burn with OMS engines was calculated by a closed form solution following main engine cutoff. Although the resulting trajectories are not optimum they are adequate to determine accurate differences between the concepts analyzed.

The FLYBACK system calculates the number of turbofan engines, fuel weight, and total flyback system weight in the booster vehicle. This routine uses the conditions at staging to estimate these quantities.

Design Optimization

Design optimization was required to enable valid comparison of the different propulsion systems. The objective was to determine the best designed vehicle for each propellant/engine type, and then compare these vehicles with each other in order to avoid any misleading results which could occur if a suboptimal design for one propellant was compared with a closer to optimal design for another.

Figure 3 diagrams the process used in the BPVIS study to optimize each vehicle design. The first step was to decide which computer variables would be fixed and which would be optimized. Certain variables like number of crew (2 for single-stage-to-orbit (SSTO), none for two-stage vehicles), number of directional control surfaces (2), number of SSME's in the recoverable P/A module of the orbiter element of the two stage vehicle (4), were held constant throughout the study.

Figure 4 summarizes the independent and some of the dependent variables used in the optimization process. This process requires that study limits be defined for each of the independent variables. A routine in HAVCD called "Design Selector" uses the range limit of each independent variable and the method of orthogonal Latin squares to define specific designs to be evaluated with the HAVCD program. These designs have independent variable values uniformly distributed in the "design space". The main feature of this technique is that a minimal number of designs have to be run on the HAVCD program. The computer time and cost savings is evident when one considers that a traditional carpet-plot

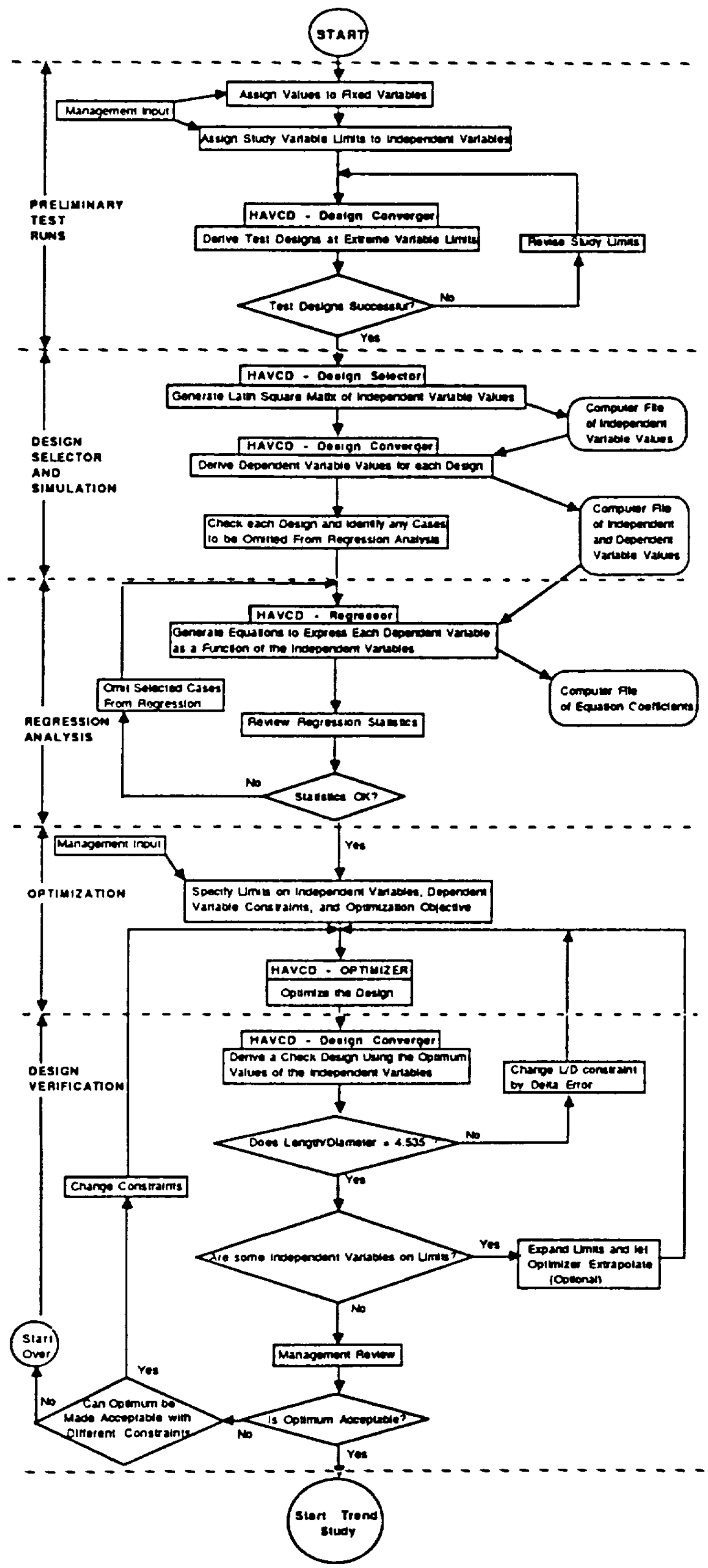


FIGURE 3. BPVIS DESIGN SYNTHESIS LOGIC FLOW

approach would require 65,536 designs to be evaluated for eight variables. Only 81 designs are required using the Latin squares method for the same number of variables. At about 20 minutes to derive a design on a VAX 8300 computer, the time savings is substantial.

Independent Variables	Application			
	Hydrogen fuel		Hydrocarbon fuel	
	2-stage	SSTO	2-stage	SSTO
Body diameter	√	√	√	√
Liftoff thrust/weight with one engine out	√	√	√	√
Booster engine mixture ratio	√	√	√	√
Number of booster engines	√	√	√	√
Booster engine nozzle expansion ratio	√	√	√	√
Orbiter propellant at staging	√	—	√	—
Booster engine mixture ratio	√	√	—	—
Second engine nozzle expansion ratio	—	√	—	√
Percent of propellant on-board at mixture ratio change	√	√	—	—
Percent of propellant on-board at booster engine shutdown	—	—	—	√
Percent of propellant on-board at expansion ratio change	—	√	—	√

Dependent Variables
Total propellant weight
Total dry weight
Propellant weight in each vehicle (Two-Stage)
Dry weight of each vehicle (Two-Stage)
Gross liftoff weight
Length/diameter ratio of booster
Booster engine weight
Booster engine vacuum specific impulse at liftoff
Total length
Propellant mass fraction
Weight at main engine cutoff
Staging velocity
Ratio of nozzle/atmospheric pressure at expansion ratio change
Engine rated thrust
Delivered booster thrust at liftoff

Figure 4 Independent and Dependent Study Variables

After the specific designs are evaluated with the HAVCD Design Converger , a multivariable regression analysis is used to fit a second order equation to the data. Each dependent variable is expressed as a function of the independent variables.

With the HAVCD optimizer, once the equations are obtained, an optimization can be performed in under ten seconds. Any of the dependent variables can be optimized or constrained to a value by the user.

STUDY RESULTS

Six single-stage configurations (designated 1A through 1F) and thirteen two-stage configurations (designated 2A through 2M) were developed in the study. Each configuration has a different type of engine. Initially, single-stage and two-stage baseline vehicles, designated 1A and 2A respectively, using SSME's were developed for comparison to subsequent optimized designs. At the conclusion of the study, it was decided to use the optimized hydrogen fuel vehicles (1B and 2B) as the reference configuration since it appeared to be more meaningful to compare the LOX/Hydrocarbon optimized designs with the LOX/Hydrogen optimized designs.

Two Stage Dry Weight Optimization

The total dry weights and booster dry weights for the two-stage configuration are shown in figure 5. Figure 6 compares these configurations to an optimized minimum dry weight LOX/LH2 configuration (2B). The dry weights obtained reflect different fuels and/or coolants and variations in vehicle size and number of engines. All of the optimized (for minimum dry weight) hydrocarbon fueled options showed an improvement over the optimized LH2 fueled concept because of the large volumetric storage requirements for LH2. The most favorable propellant for minimum dry weight propellant is the methane-fueled, LH2-cooled concept (2G). Methane has the advantage of good specific impulse and appreciable fuel bulk density.

Configuration	2.A	2.B	2.C	2.D	2.E	2.F	2.G	2.H	2.I	2J	2.K	2.L	2.M
Fuel	H_2	H_2	RP-1	RP-1	RP-1	RP-1	CH_4	CH_4	NBP C_3H_8	NBP C_3H_8	SC C_3H_8	SC C_3H_8	SC C_3H_8
Coolant	H_2	H_2	H_2	H_2	RP-1	RP-1	H_2	CH_4	H_2	NBP C_3H_8	H_2	NBP C_3H_8	C_3H_8
Mixture Ratio	6:1	8.97:1	3.26:1	3.15:1	3.15:1	2.5:1	3.77:1	3.7:1	3.09:1	3.42:1	3.42:1	3.35:1	3.44:1
Number of Booster Engines	7	5	5	6	5	5	5	5	6	5	5	5	5
Booster Engines Vac. Thrust (lb)	494,000	661,400	656,340	690,530	675,400	855,660	596,070	690,740	545,710	740,560	653,940	766,010	734,680
P_C (psia)	3,270	4,000	4,000	2,500	675,400	1,650	4,300	3,300	4,000	2,600	4,000	3,300	3,900
Vacuum Isp - sec	437	416	326	311	294	304	347	338	328	316	330	318	325
Nozzle Expansion Ratio	35.0	50.3	28.8	45.0	15.0	15.0	22.7	28.9	21.6	22.8	25.0	28.2	29.9
N Near-Term F Far-Term	N	N	N	N	N	F	N	N	N	N	N	N	F
Booster Dry Weight (lb)	241,720	196,610	167,630	171,620	190,500	187,330	159,150	167,130	163,480	170,590	165,280	171,980	166,720
Orbiter Dry Weight (lb)	163,420	164,030	164,150	164,410	163,310	162,610	163,450	163,830	163,470	164,870	163,470	163,710	164,780
Total Dry Weight (lb)	405,140	360,630	331,780	336,030	353,810	349,940	322,600	330,960	326,950	335,460	328,750	335,690	331,490
GLOW (lb)	3,167,600	3,341,800	3,469,800	3,609,300	3,934,400	3,569,300	3,289,100	3,564,200	3,336,700	3,731,900	3,353,700	3,541,100	3,593,600
$V_{Staging}$ (ft/s)	5,000.0	4,922.3	4,232.2	4,172.7	5,278.2	5,075.1	4,743.8	5,135.6	4,425.3	4,280.9	4,518.5	4,624.8	4,180.6

Figure 5 Two-Stage Vehicle Optimized Results

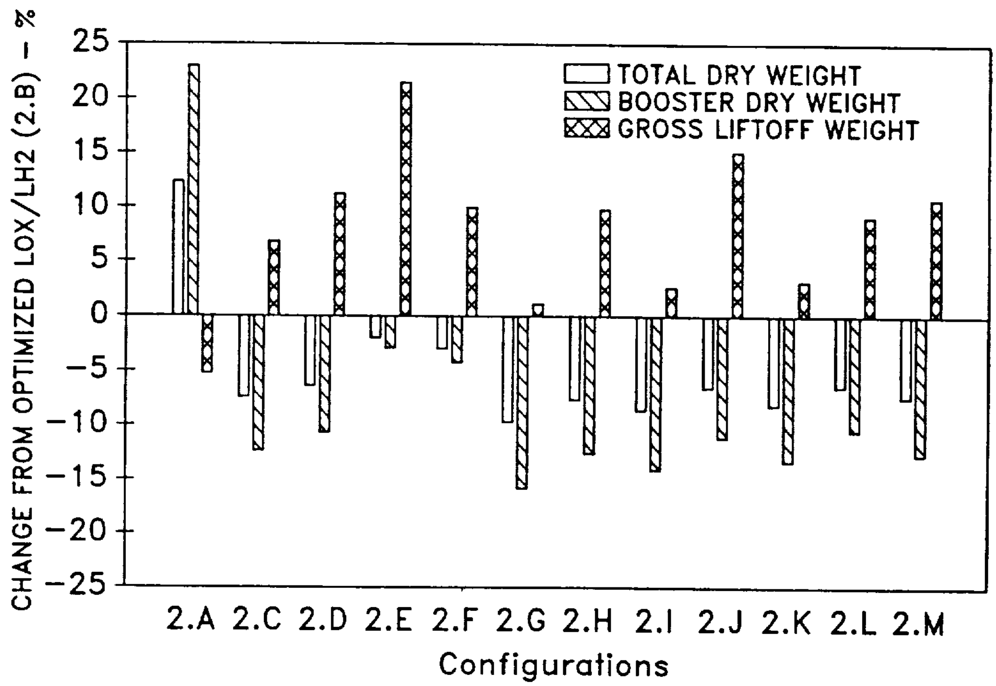


Figure 6. TWO STAGE WEIGHT COMPARISONS

Also shown in figure 6, there is no significant improvement in dry weight of subcooled propane compared to normal boiling point propane.

All hydrocarbon fuels produced an increased gross liftoff weight (GLOW) relative to the optimized LOX/LH2 system. RP-1 propellant with RP-1 cooling for a near term engine caused the greatest GLOW. Methane propellants, with hydrogen coolants, produced the smallest increase in GLOW. Hydrogen cooling is of greatest benefit for the RP-1 fuel engines (6% lower dry weight) and may not be worth its expense and added complexity for methane and propane fuels (2.5% lower dry weight).

Two Stage Optimized Designs

Three optimized results are presented in figures 7,8, and 9. The optimized LOX/hydrogen vehicle using Aerojet engines are shown in Figure 7. Notice that the flyback booster uses a tapered hydrogen tank to accommodate an aerodynamic shape. Figure 8 shows the optimized vehicle of a low chamber pressure RP-1 engine. Note that the base area requires a body flare to accommodate the large rocket engines. The lightest dry weight vehicle (methane tri-propellant) is shown in figure 9.

Configuration: 2.B

Booster:

Weights:	
Dry Weight (lb) =	197,470
Propellant Weight (lb) =	1,074,000
- LO_2 (lb) =	966,380
- LH_2 (lb) =	107,690
Inert Weight (lb) =	227,380
λ' =	0.819

Engines:	
Type:	LH_2/LO_2
Number =	5
Thrust (vacuum, each) (lb) =	671,110
MR:	8.97
P_C (psia) =	4,000
I_{sp} =	416
ϵ =	50.3
$d_{powerhead}$ (in) =	108.0
D_{nozzle} (in) =	74.0

Body:	
$\frac{\ell}{D}$ =	4.53
D (ft) =	30.5
$S_{body\ flap}$ (ft^2) =	244

Fins:	
S_F (ft^2) (ea) =	144
AR =	1.39
λ =	0.55
t/c =	11%
S_{rudder} (ft^2) (ea) =	43.3

Wing:	
S_{ref} (ft^2) =	3,132
AR =	2.06
λ =	0.11
t/c =	11%
$S_{flaperons}$ (ft^2) =	626

Flyback Engines:	2

Orbiter:

Weights:	
Dry Weight (lb) =	164,380
Propellant Weight (lb) =	1,601,000
- LO_2 (lb) =	1,372,000
- LH_2 (LB) =	228,670
Inert Weight (lb) =	192,050
λ' =	0.893

P/A Module (4 SSMEs):	
Weight (lb) =	122,000
Circularization OMS Propellant (lb) =	9,470
Total OMS Propellant (lb) =	18,600

GLOW (lb) =	3,253,700
$V_{staging}$ (ft/s) =	4,524
P/L to Space Station (lb) =	150,000

- Optimized LOX/LH_2
- Single relatively high MR ($\approx$9:1)

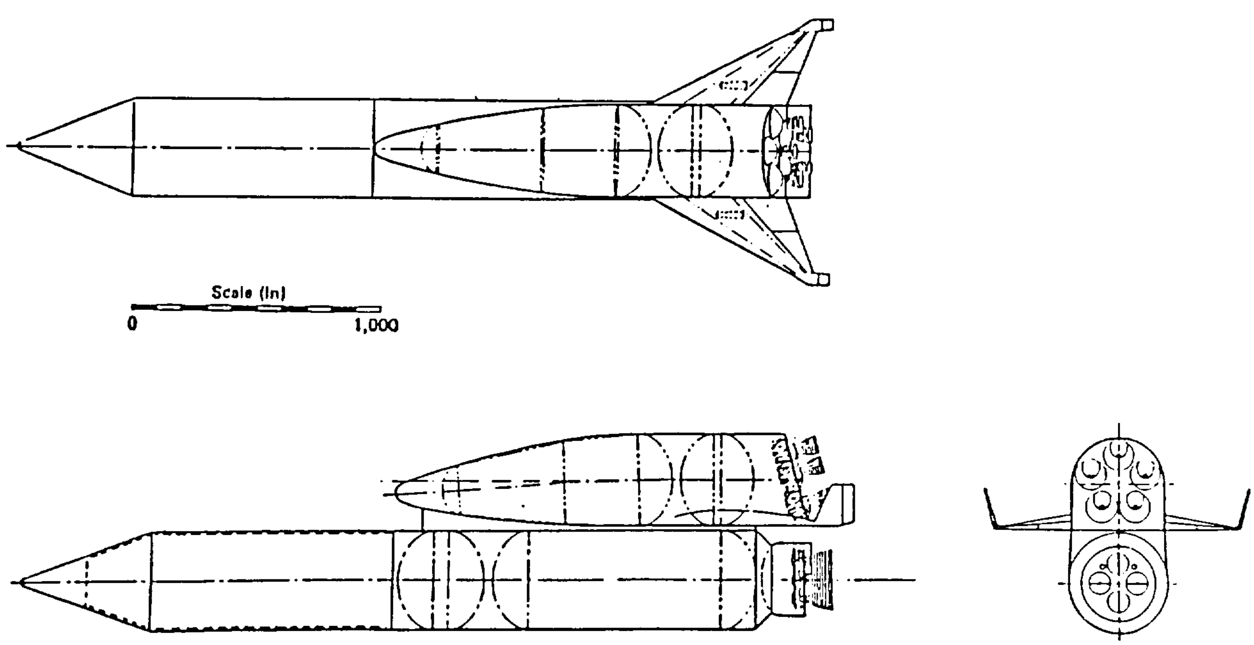


Figure 7. Configuration 2.B Summary

Configuration: 2.E

Booster:

Weights:	
Dry Weight (lb) =	191,000
Propellant Weight (lb) =	1,865,000
- LO_2 (lb) =	1,415,000
- RP-1 (lb) =	449,000
Inert Weight (lb) =	199,000
λ' =	0.888

Engines:	
Type:	RP-1/RP-1
Number =	6
Thrust (vacuum, each) (lb) =	620,000
MR:	3.15
Pc (psia) =	1,300
I_{sp} =	294
ε =	15.00
$d_{powerhead}$ (in) =	120.0
D_{nozzle} (in) =	69.4

Body:	
$\frac{\ell}{D}$ =	4.53
D (ft) =	27.2
$S_{body\ flap}$ (ft^2) =	218

Fins:	
S_F (ft^2) (ea) =	141
AR =	1.39
λ =	0.55
t/c =	11%
S_{rudder} (ft^2) (ea) =	42.23

Wing:	
S_{ref} (ft^2) =	3,023
AR =	2.06
λ =	0.11
t/c =	11%
$S_{flaperons}$ (ft^2) =	605

Flyback Engines:	2

Orbiter:

Weights:	
Dry Weight (lb) =	163,000
Propellant Weight (lb) =	1,493,000
- LO_2 (lb) =	1,280,000
- LH_2 (LB) =	213,000
Inert Weight (lb) =	190,000
λ' =	0.887

P/A Module (4 SSMEs):	
Weight (lb) =	122,000
Circularization OMS Propellant (lb) =	9,470
Total OMS Propellant (lb) =	18,600

GLOW (lb) =	3,934,000
$V_{staging}$ (ft/s) =	5,278
P/L to Space Station (lb) =	150,000

- Optimized LOX/RP-1
- RP-1 engine cooling at low Pc = 1300°F (no film cooling)
- Acceptable engine section "flare"

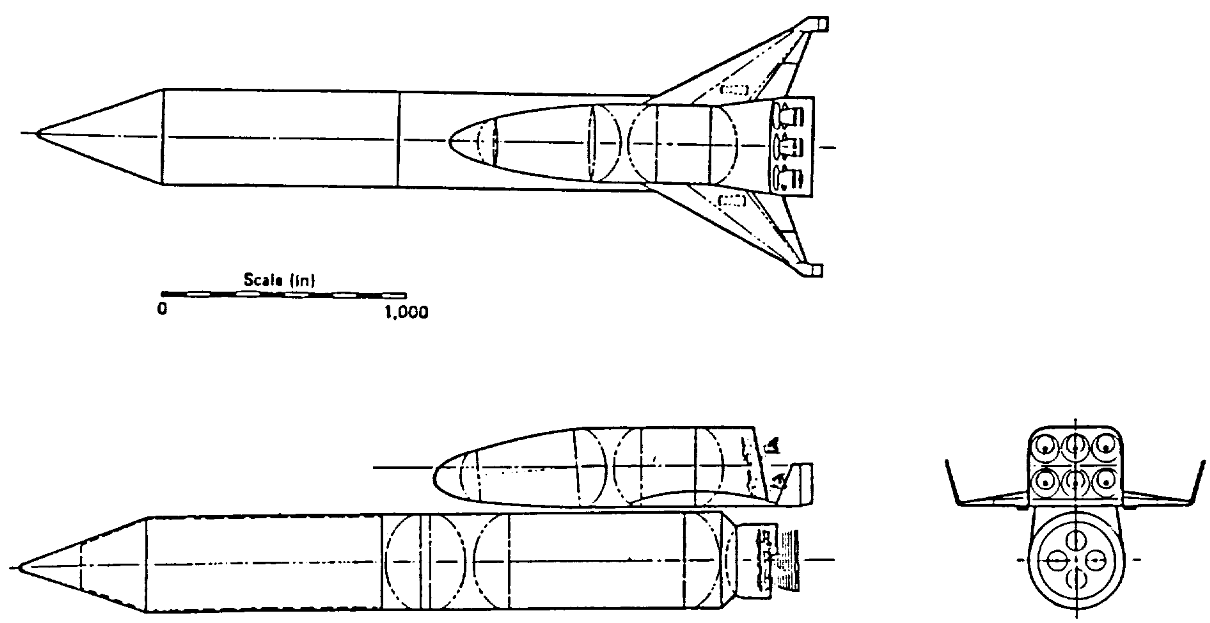


Figure 8. Configuration 2.E Summary

Configuration: 2.G

Booster:

Weights:	
Dry Weight (lb) =	159,150
Propellant Weight (lb) =	1,244,000
- LO_2 (lb) =	983,000
- Methane (lb) =	238,000
- LH_2 (lb) =	22,900
Inert Weight (lb) =	188,000
λ' =	0.863

Body:	
$\frac{\ell}{D}$ =	4.53
D (ft) =	30.3
$S_{body\ flap}$ (ft^2) =	243

Wing:	
S_{ref} (ft^2) =	2,245
AR =	2.36
λ =	0.11
t/c =	11%
$S_{flaperons}$ (ft^2) =	449

Engines:	
Type:	Methane/LH_2
Number =	5
Thrust (vacuum, each) (lb) =	596,000
MR:	4.13
P_c (psia) =	4,300
I_{sp} =	347
ε =	22.75
$d_{powerhead}$ (in) =	92.7
D_{nozzle} (in) =	44.6

Fins:	
S_F (ft^2) (ea) =	125
AR =	1.39
λ =	0.55
t/c =	11%
S_{rudder} (ft^2) (ea) =	37.6

Flyback Engines:	2

Orbiter:

Weights:	
Dry Weight (lb) =	163,000
Propellant Weight (lb) =	1,507,000
- LO_2 (lb) =	1,292,000
- LH_2 (lb) =	215,000
Inert Weight (lb) =	191,000
λ' =	0.888

P/A Module (4 SSMEs):	
Weight (lb) =	122,000
Circularization OMS Propellant (lb) =	9,470
Total OMS Propellant (lb) =	18,600

GLOW (lb) =	3,289,000
$V_{staging}$ (ft/s) =	4,734
P/L to Space Station (lb) =	150,000

- Optimized LOX/CH_4
- Separate LH_2 tank for engine cooling

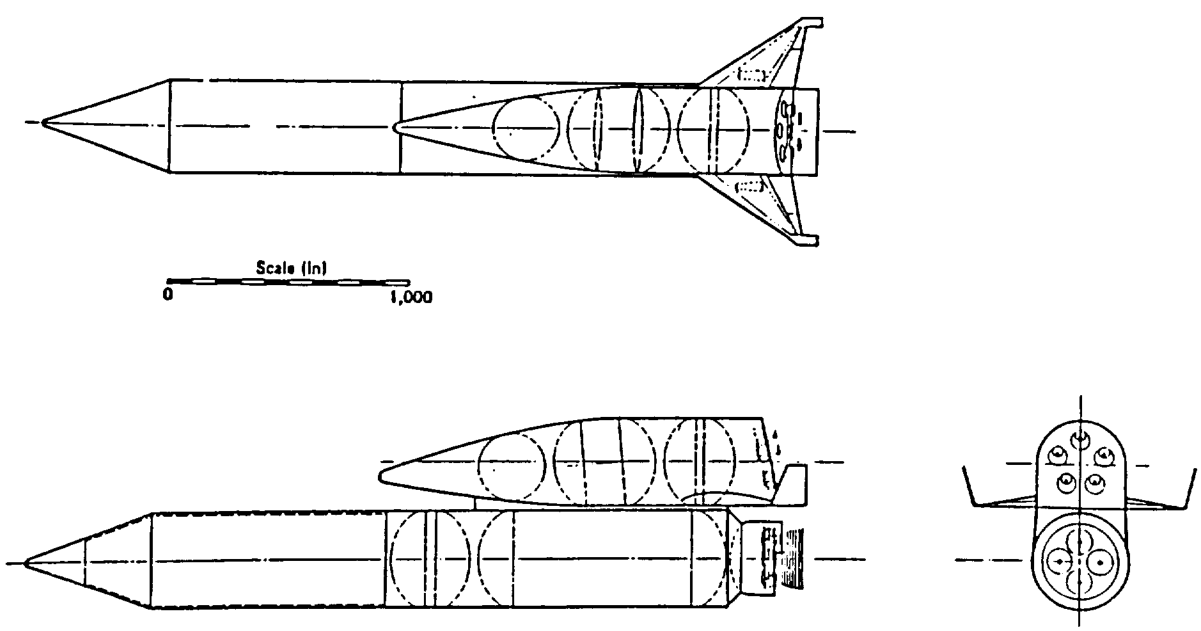


Figure 9. Configuration 2.G Summary

Two Stage Expansion Ratio Change Sensitivities

Included in this study was an evaluation of changing the booster engine nozzle to a higher expansion ratio at some point in the boost phase. Four configurations were evaluated, LOX/LH2, LOX/RP-1 (H2 cooled), LOX/RP-1 (RP-1 cooled), and LOX/Methane (H2 cooled). The liftoff nozzles positions were set at 30:1, 15:1, 15:1 and 15:1 expansion ratios respectively. Expansion ratios were change to 80 or 100:1 over an altitude range of 10,000 to 70,000 ft. were evaluated. It was found that dry weight increased when an extension nozzle is used at any altitude (Figure 10). Total dry weight was minimized with the booster engines at constant expansion ratio during boost, set at a low ratio.

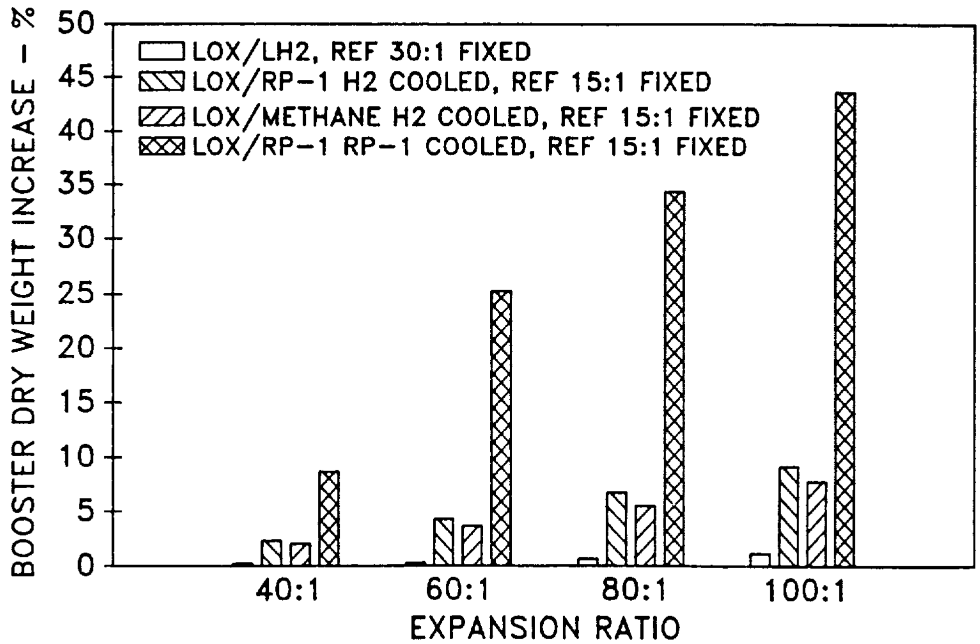


FIGURE 10 EXTENDED NOZZLE EXPANSION RATIO IMPACT ON TWO STAGE BOOSTER DRY WEIGHT

Two Stage Chamber Pressure Sensitivities

Booster engine chamber pressure was evaluated to determine its influence on dry weight for LOX/LH2, RP-1, and Methane fuels. Since vehicle aft body size, as well as engine and nozzle weight, in general, decreases with increased chamber pressure, dry weight is minimized at the higher chamber pressure (Figure 11).

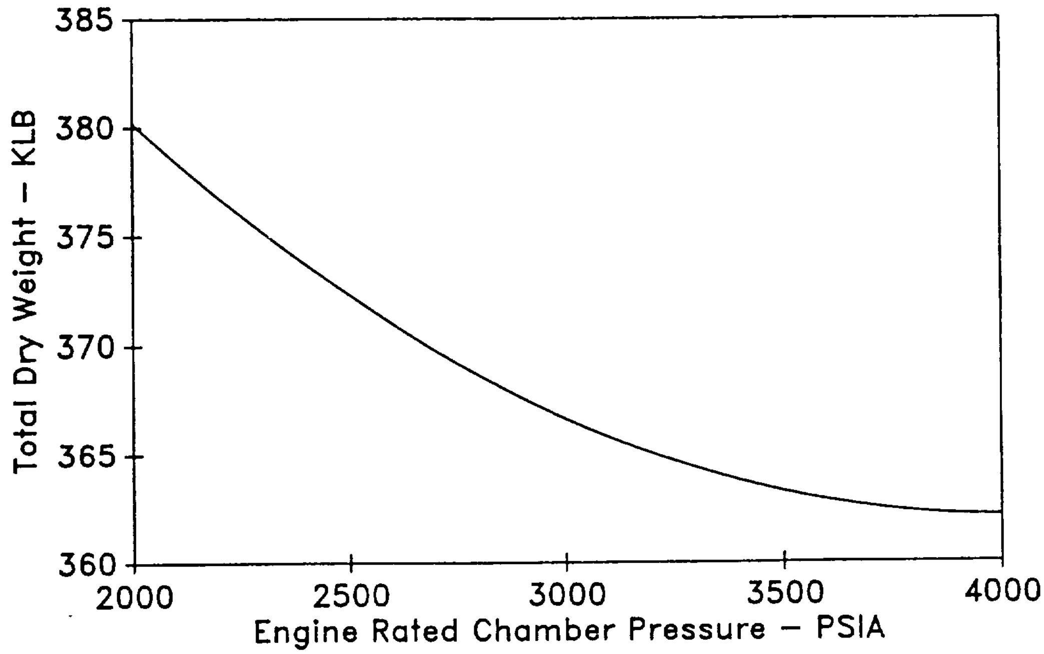


FIGURE 11 TYPICAL TWO STAGE DRY WEIGHT SENSITIVITY TO CHAMBER PRESSURE

Two Stage Variable/High Mixture Ratio LOX/LH2 Evaluation

Changing the LOX/LH2 variable mixture ratio range during the boost phase from a high (8-18:1) range to a lower (6-12:1) range was investigated to improve propellant bulk density and system efficiency. It was assumed that mixture ratio would be changed by changing the oxidizer flow rate only while maintaining a constant hydrogen flowrate. Consequently, chamber pressure and engine thrust are reduced by the mixture ratio reduction. It was found that specific impulse improvement during the flight had little effect on minimizing the booster dry weight. Improvement in bulk density had a more significant effect of reducing dry weight. For example, increasing a single mixture ratio from 6.00:1 (for maximum specific impulse) to about 9.00:1 produced a lower dry vehicle weight. A single Booster mixture ratio was also evaluated. The LOX/LH2 configuration was optimized to a single mixture ratio of 8.97. The dry weight increased by only 1.5% when a single mixture ratio is used compared to the use of a more complex variable mixture ratio of 12.0 reducing to 6.0. It was therefore concluded that variable mixture ratio LOX/LH2 main engines do not provide a significant payoff.

Crossfeed Evaluation

Cross-feeding propellant from the first-stage propellant tanks to the second stage engines during the boost phase was evaluated. Line diameter on the first-stage must be increased to accommodate the increased flow rates, and relatively heavy prevalves, plumbing and structure must also be added. It was found that with the relatively low staging velocity of the booster, implementing crossfeed to reduce dry weight is not effective. Figure 12 summarizes the effect of crossfeed on launch vehicle design by comparing the weight of configuration 2B with crossfeed to the same configuration without crossfeed at two mixture ratios.

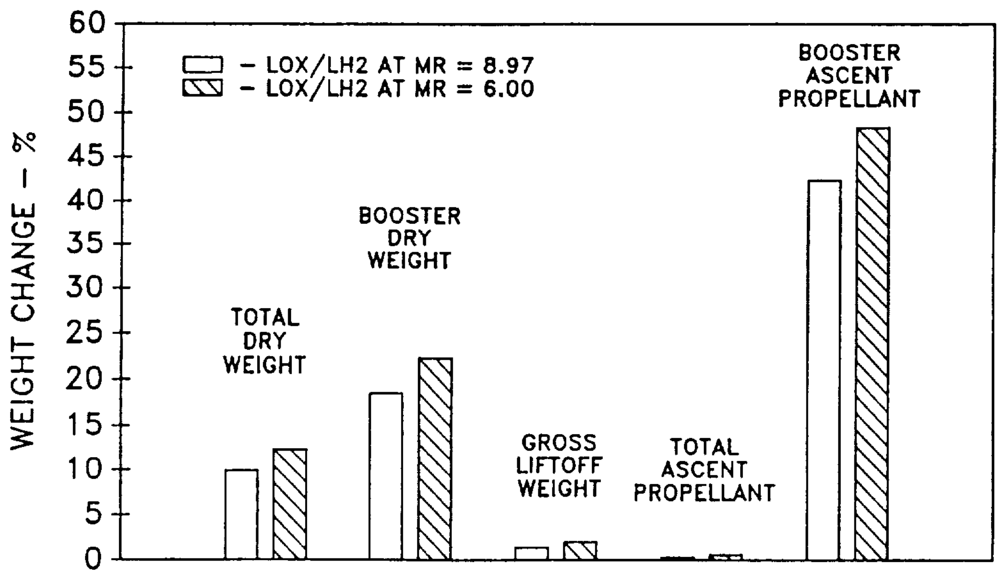


FIGURE 12 EFFECT OF CROSSFEED ON VEHICLE DESIGN

Two Stage Near Term versus Far Term

The benefit of greater chamber pressures and higher specific impulse for RP-1 and Propane fuel engines is shown in figure 13. As shown, the benefit is small and probably not worth the expenditure of resources in this area.

SSTO Computer Model Comparison

The Boeing SSTO model results for LOX/LH2 was compared to Reference 4 study (Figure 14). Different payloads, orbit inclination, and other assumptions between the two models required that both results be normalized for direct comparison. A fair agreement exists between the two models with Boeing's model being the more conservative of the two in dry weight determination.

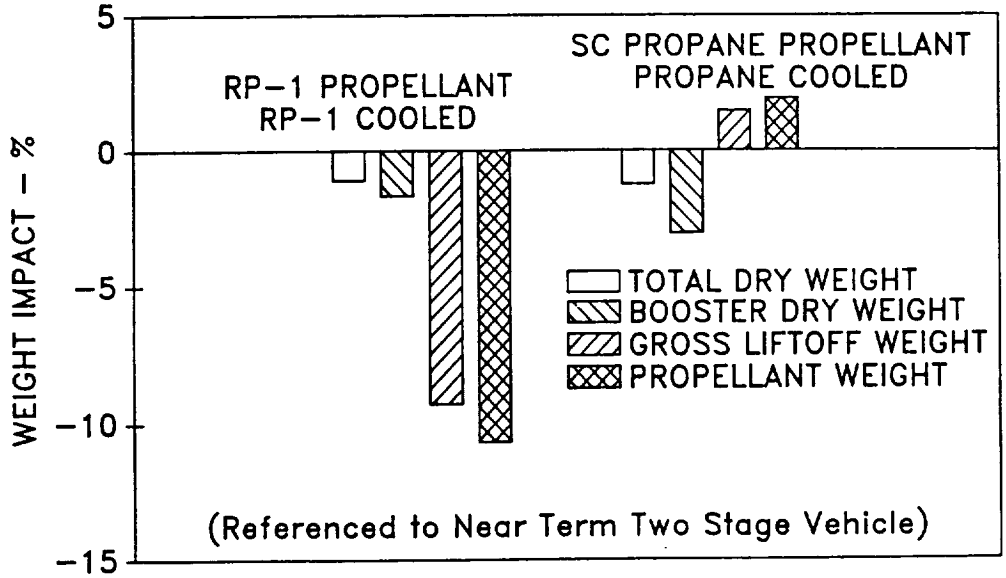


FIGURE 13 FAR TERM TECHNOLOGY IMPACT ON SYSTEM WEIGHTS

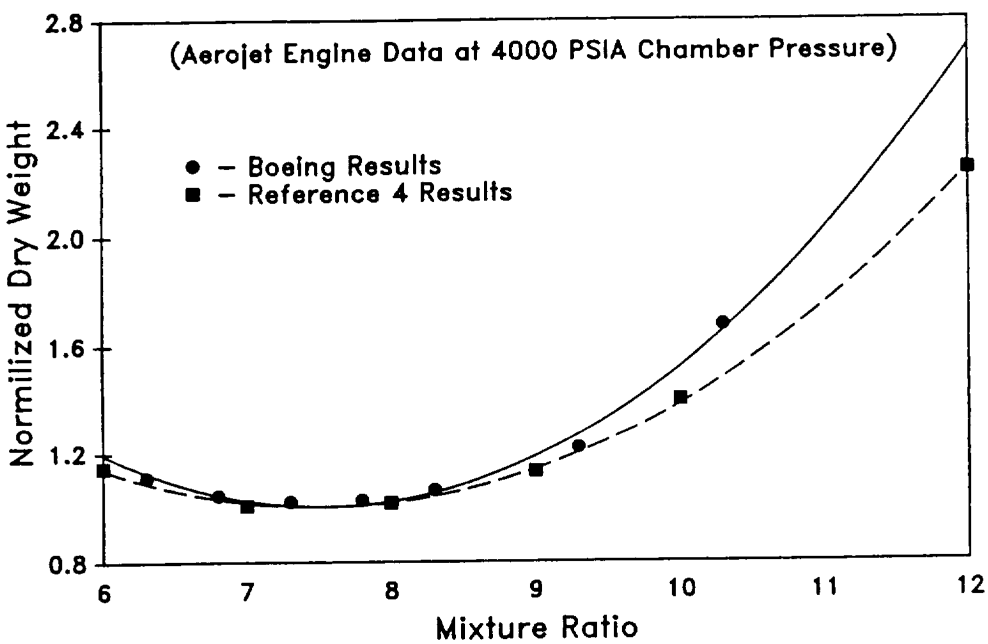
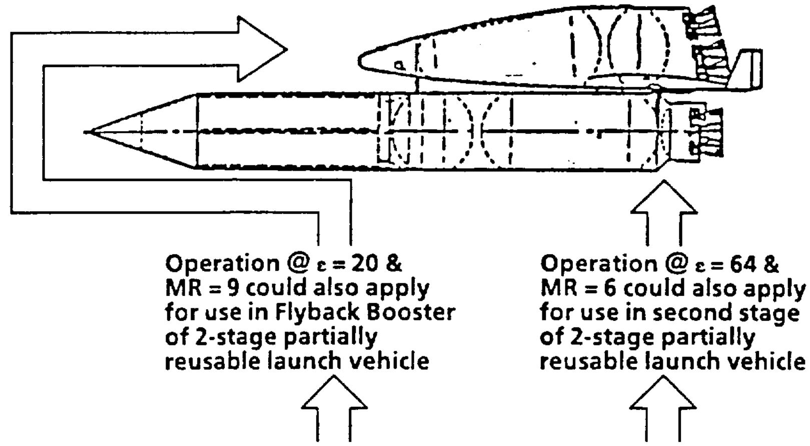


FIGURE 14 BOEING NORMALIZED LOX/LH2 SSTO STUDY RESULTS COMPARED TO REF. 4 NORMALIZED STUDY RESULTS

<u>Full Flow Topping Cycle, Variable MR Engine Impact</u>

The potential benefit of a full flow topping cycle, variable mixture ratio, variable expansion ratio engine was also assessed for the SSTO. The features of this engine, as projected by Acurex Corp., under subcontract to Boeing, are summarized in figure 15. The mixture ratios and expansion ratios were chosen by Acurex and are not necessarily the optimum values.



AREA RATIO	20	64	64
THRUST (VAC), LBS	679,952	700,000	523,225
MIXTURE RATIO, O/F	9	9	6
CHAMBER PRESSURE, PSIA	3,000	3,000	2,250
AREA THROAT, SQ. IN.	125.6	125.6	125.6
DIAMETER THROAT, IN.	12.64	12.64	12.64
DIAMETER EXIT, IN.	56.6	101.2	101.2
WEIGHT FLOW RATE, OXIDIZER, LB/SEC	1,485	1,485	990
WEIGHT FLOW RATE, FUEL, LB/SEC	165	165	165
TOTAL WEIGHT FLOW, LB/SEC	1,650	1,650	1,155
SPECIFIC IMPULSE (VAC), SEC	412	424	453
ENGINE DRY WEIGHT WITH NSI, LBS	6,860	---	---
ENGINE DRY WEIGHT WITHOUT NSI, LBS	---	6,565	6,565
ENGINE THRUST-TO-WEIGHT RATIO	83	107	79.7
ENGINE LENGTH, IN.	154	154	154
DIAMETER POWER HEAD, IN.	100	100	100

Figure 15. Acurex SSTO Engine Characteristics

SSTO Dry Weight Factor Evaluation

The dry weight factor is a technology multiplier that reduces across the board all dry weight components which is needed to make an SSTO feasible. This study used a 25% weight reduction (dry weight factor of 0.75) to provide a GLOW around 1.25 million pounds. If GLOW is allowed to increase, the dry weight factor can be increased to about 0.92 for subcooled propane, about 0.91 for methane, and about 0.87 for the Acurex hydrogen engines. All dry weight factors peaked at a GLOW of about 3.5 million pounds (Figure 16). These plotted values have not been optimized and further improvement is expected in the dry weight when each configuration is optimized at each GLOW. This is planned for future accomplishment on IR&D funding. The non-optimum values used to create the curve of Figure 16 is tabulated in Figure 17. The non-optimized dry weight is included in the table.

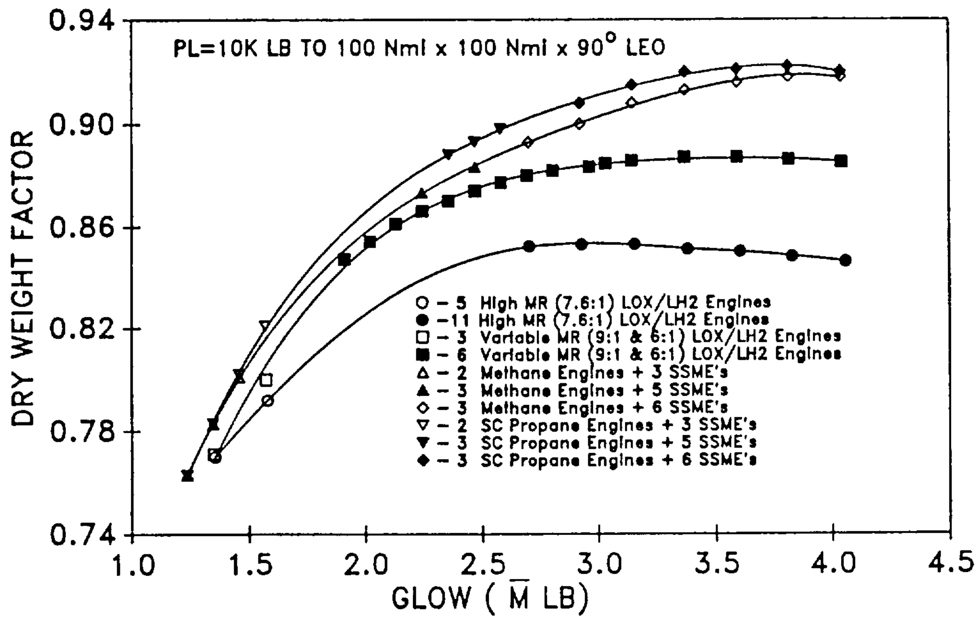


FIGURE 16 DRY WEIGHT FACTOR FOR SSTO VEHICLE CONCEPTS

Fixed Mixture Ratio				Variable Mixture Ratio LOX/LHZ			
Glow x 10⁶	Dry Weight	Dry Wt. Factor	Engines	Glow x 10⁶	Dry Weight	Dry Wt. Factor	Engines
1.353	119,760	.770	5	1.348	114,740	.771	3
1.578	141,680	.792	5	1.572	157,210	.800	3
2.705	250,970	.852	11	1.909	167,750	.847	6
2.930	273,030	.853	11	2.021	178,350	.854	6
3.156	295,090	.853	11	2.133	188,920	.861	6
3.381	317,190	.851	11	2.246	199,530	.866	6
3.607	339,320	.850	11	2.358	210,120	.870	6
3.832	361,440	.848	11	2.470	220,710	.874	6
4.059	385,290	.848	11	2.583	231,310	.877	6
				2.695	241,910	.880	6
				2.807	252,510	.882	6
				2.963	263,120	.883	6
				3.032	273,710	.885	6
				3.144	284,000	.886	6
				3.368	305,600	.887	6
				3.593	326,800	.887	6
				3.818	348,100	.886	6
				4.042	369,400	.885	6

LOX/Methane				LOX/SC Propane			
Glow x 10⁶	Dry Weight	Dry Wt. Factor	Engines HC/SSME	Glow x 10⁶	Dry Weight	Dry Wt. Factor	Engines HC/SSME
1.235	105,200	.763	2/3	1.235	105,110	.765	2/3
1.345	113,560	.783	2/3	1.344	113,300	.784	2/3
1.455	122,470	.801	2/3	1.455	122,270	.802	2/3
2.245	204,300	.873	3/5	1.568	133,610	.821	2/3
2.470	221,200	.883	3/5	2.360	212,920	.888	3/5
2.702	250,000	.893	3/6	2.470	221,210	.893	3/5
2.923	267,700	.900	3/6	2.581	230,900	.898	3/5
3.147	288,400	.908	3/6	2.923	267,800	.908	3/6
3.371	308,700	.913	3/6	3.147	288,100	.915	3/6
3.594	328,000	.916	3/6	3.371	307,900	.920	3/6
3.817	348,000	.918	3/6	3.593	327,200	.921	3/6
4.040	367,000	.918	3/6	3.815	345,900	.922	3/6
				4.037	364,100	.920	3/6

Figure 17. Table for SSTP Dry Weight Factors

SSTO Dry Weight Optimization

The optimized SSTO configurations for total dry weight are shown in Figure 18. Figure 19 compares the hydrocarbon configurations to an optimized, for minimum dry weight, LOX/LH2 configuration. For comparative purposes, all of the configurations were optimized with a dry weight factor of 0.75. The hydrocarbon configurations show up to a 5% reduction in dry weight over the optimized LOX/LH2 configuration. The improved propellant bulk density of the hydrocarbons improve both the dry weight and GLOW for methane and subcooled propane.

Configuration	1.A	1.B	1.C	1.D	1.E	Acurex
Fuel	LH_2	LH_2	RP-1	Methane	SC Propane	LH_2
Coolant	LH_2	LH_2	LH_2	LH_2	LH_2	LH_2
Mixture Ratio	6.0	7.6	3.03	4.19	3.59	9.6
Number of Main Engines	5	5	2[1]	2[1]	2[1]	3
Main Engines Vac. Thrust (lb)	504,120	381,440	338,240	280,622	279,670	775,570
Vacuum Isp - sec	448	425	312	329	317	424/453
Booster P_C (psia)	3,270	4,000	4,000	4,300	4,000	3,000/2,250
Expansion Ratio	55/150	30/100	15[2]	15[2]	15[2]	20/64
Propellant Remaining @ Main Engine Cutoff	N/A	N/A	30%	37%	42%	N/A
Main Engine Total Thrust Range Ratio	.25	.25	.82	.96	Not Done	Not Done
SSME Engine Total Thrust Range Ratio	N/A	N/A	.41	.39	Not Done	N/A
Inert Weight Factor	.75	.75	.75	.75	.75	.75
Dry Weight	141,020	104,690	102,080	100,040	99,216	103,460
Propellant	1,283,000	1,062,300	1,130,300	1,039,200	1,029,400	1,092,400
Glow	1,460,000	1,119,750	1,263,600	1,168,200	1,157,400	1,226,700

(1) Plus 3 SSME Engines.
(2) Initial Expansion Ratio 55 changed to 150 on SSME Engines.

Figure 18. Single-Stage Optimized Results

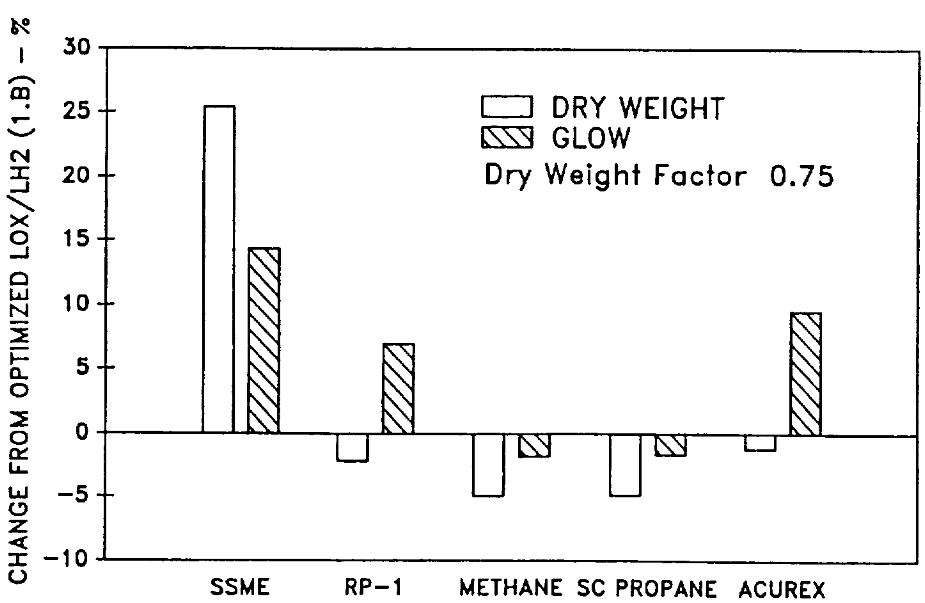


FIGURE 19 SINGLE STAGE WEIGHT COMPARISONS

SSTO LOX/LH2 Variable Mixture Ratio Impact

Allowing the mixture ratio to change during the ascent of the LOX/LH2 SSTO vehicle was found to generate a minimum dry weight system. Liftoff mixture ratio optimized at 8.4:1 and second mixture ratio optimized at about 7.5:1. However, the optimized variable mixture ratio system is less than 2% lighter in dry weight than a fixed mixture ratio system which optimized at 7.6:1.

SSTO LOX/LH2 Variable Expansion Ratio Impact

The all LOX/LH2 SSTO optimized at a liftoff expansion ratio of 30:1 and the second expansion ratio in 100:1. The LH2 plus hydrocarbon fueled SSTO used an expansion ratio change on the LOX/LH2 engines of 55:1 at liftoff, changing to 100:1 later in the trajectory. All the hydrocarbon engines optimized at the lowest allowed expansion ratio (15:1) to minimize system dry weight.

CONCLUSIONS

Two stage and single stage vehicle dry weights can be minimized by using hydrocarbon propellants. Methane should provide a minimum dry weight of 10.5% less than an optimized LOX/hydrogen two stage vehicle. Subcooled propane will provide a 5.2% reduction in single stage vehicle dry weight when compared to an optimized LOX/hydrogen vehicles at a dry weight factor of 0.75.

Two Stage

For minimum dry weight, two stage booster benefits with high chamber pressure engines. Extended nozzle for the booster engines is not beneficial plus low expansion ratio fixed nozzles proved to minimized dry weight.

Variable mixture ratio for a LOX/hydrogen proves to be of little benefit for minimum dry weight. However, a higher mixture ratio (8.97:1) than for highest specific impulse (6.0:1) improves minimum dry weight. Crossfeeding propellants shows no dry weight benefit. Engine development to increase chamber pressure and performance has slight dry weight improvement.

Optimum staging velocity of the two-stage, parallel-burn, partially reusable heavy lift vehicles for minimum dry weight is about 5000 feet per second. The high propellant mass fraction of the second stage drove the staging velocity regardless of engine type or propellant on the booster. The partially reusable second stages of all vehicle options examined are all nearly identical.

Except for two stage boosters using LH2 engine cooling and either methane or normal boiling point propane as fuel, all engine types and propellant combinations appear to allow boosters not to require canards. The wing/body aerodynamics for adequate controllability throughout the flight regime, in most cases, is adequate without the use of canards.

The most effective flyback booster engine for minimizing dry weight of parallel-burn two-stage partially reusable vehicle a gas-generator, LOX/methane/LH2 cooled approach having as high a combustion chamber pressure as practical. However, this tri-propellant system may require a canard because of the increased booster length for the coolant tank, the weight of the canard was included in this configurations weight.

LOX/methane with methane cooling is slightly heavier (8,360 LBS) then methane with hydrogen cooling. This system is less complex an would not require a canard.

The use of LOX/RP-1 with LH2 cooling avoids the cryogenic handling and storage of the other fuels. This propellant combination increases dry weight only about 12,500 lb compared to the methane tri-propellant and would not require a flyback booster canard.

If LOX/hydrocarbon engine required for low staging velocity flyback booster, the best compromise engine may be the low development cost, gas-generator LOX/RP-1 with RP cooling at a low chamber pressure (1300 psia without film cooling). Only a total of two cryogen tanks would be required and the booster would not require a canard.

The LOX/propane options (normal boiling point or subcooled) offer sufficient advantages over RP-1 or methane to warrant further consideration. The LOX/propane with propane cooling appears to be the best compromise among the propane fueled options considered. The normal boiling point propane with LH2 cooling was only slightly heavier (1,200 LBS) then the subcooled tri-propellant, but avoids sub-cooling and the added facility requirements.

Single Stage

Variable mixture shows some benefit to the dry weight of an all LOX/LH2 SSTO vehicle. However, fixed mixture ratio of 7.6:1 has only a 2% dry weight penalty for a simpler engine at a dry weight factor of 0.75.

Increasing GLOW to allow a smaller dry weight factor, appears to be an effective strategy to allow lower risk development of an SSTO vehicle and should be evaluated further (IR&D result).

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