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DESIGN OPTIMIZATION OF GAS GENERATOR HYBRID PROPULSION BOOSTERS*

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ABSTRACT

This paper presents a methodology used in support of a contract study for NASA/MSFC to optimize the design of gas generator hybrid propulsion booster for uprating the National Space Transportation System (NSTS). The objective was to compare alternative configurations for this booster approach, optimizing each candidate concept on different bases, in order to develop data for a trade table on which a final decision was based. The methodology is capable of processing a large number of independent and dependent variables, adjusting the overall subsystems characteristics to arrive at a best compromise integrated design to meet various specified optimization criteria subject to selected constraints. For each system considered, a detailed weight statement was generated along with preliminary cost and reliability estimates.

INTRODUCTION

Hybrid propulsion systems have been recommended for Space Shuttle application for over eight years. In 1982, the NASA/MSFC "Shuttle Derived Vehicle Technology Requirements Study" rated hybrid propulsion technology as the highest priority of 23 technologies when ranked by economic leverage. In 1987, the NASA/LRC, "Analysis of Quasi Hybrid Booster Concepts" study recommended that future efforts for advanced earth-to-orbit booster systems focus on conventional hybrid rockets. As a result of increased interest in improving launch vehicle safety and reliability, the Aerospace Safety Advisory Panel Annual Report, March 1990, recognized the capability of hybrid rocket technology to improve Space Shuttle launch safety and reliability, and to reduce hazardous environmental conditions that result from the combustion of current solid rocket propellants. Hybrid rocket propulsion has been used in operational hybrid missiles (Sandpiper, Firebolt, HAST), and tested from idle to 75,000 lbf thrust in ground tests, but design algorithms and modeling methods need to be developed and verified with test data for space booster applications.

This paper presents the results of a conceptual design study to determine the best hybrid booster configuration for STS application (Ref 1). The study groundruled that the booster should deliver the same thrust versus time profile as the ASRM (Advanced Solid Rocket Motor). Previous studies have considered the classic hybrid rocket with a solid fuel and liquid or gaseous oxygen injected at the forward end of the fuel grain. This study not only examined the classic hybrid concept, but also a newer, gas generator concept which uses a solid propellant gas generator to provide a fuel-rich gas that is burned in a combustion chamber. This concept is similar to the ducted rocket engine except that liquid oxygen is used instead of air from the atmosphere.

The study developed four configurations of the gas generator concept and four configurations of the classic hybrid rocket concept. These configurations were comprised of pump or pressure fed engines with ~~liquid~~ oxygen or hydrogen peroxide (H2O2) oxidizer. A design program was used to size the boosters, compute payload capability, and estimate life cycle cost and reliability.

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The best configuration of these eight was selected for design optimization. Different optimized designs were derived for lowest life cycle cost, greatest payload capability, lowest cost/payload weight ratio, lowest empty weight, and lowest gross lift-off weight. A comparison of the independent and dependent variables for each design provides design insight, and provides options for booster design.

DESIGN PROGRAM

The hybrid propulsion design program was derived from the HAVCD (Hypervelocity Aerospace Vehicle Conceptual Design) program used in the BP/VIS (Booster Propulsion/Vehicle Impact Study, Ref. 2). The original code combined aerodynamic, propulsion, weight, tank sizing and pressurization, trajectory performance, and flyback system design subprograms to design single stage and two stage to orbit, rocket powered launch vehicles. The basic methodology of the program was retained, but the subprograms were modified to analyze hybrid rocket boosters. Figure 1 illustrates the different analysis programs used in the new code. The weight subprogram serves as the primary analysis routine, with iterations between it and the tank sizing, propellant weight, pressure vessels, and nose section subprograms to achieve a consistent, integrated design that matches the ASRM thrust profile, figure 2. The performance (i.e.payload capability), cost, and reliability of the design is evaluated by their respective subprograms. The optimization capability of the original program was retained and used on the one best configuration selected from the initial eight configurations. A summary of the optimization technique is described in a following section.

AERODYNAMIC MODEL

The aerodynamic subprogram uses a blend of simplified aerodynamic theory and empirical relationships which result in acceptable agreement with wind tunnel test data. It 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 launch vehicle geometry determined by the weight subprogram. The primary modification to the subprogram from the original version was to account for the interference drag between the Shuttle external tank and the hybrid boosters. It was at its maximum when the booster height was the same as the external tank, and decreased as booster length increased.

WEIGHT MODEL

The weight subprogram collects output from the other interactive subprograms. It calls the appropriate subprograms to get component size, weights, locations, and center of gravity travel. Since variables in one subprogram influence calculated variables in other subprograms, the weight subprogram cycles through all of the other subprograms until system and subsystem weights converge to a constant value. Data files are created for use by the cost, reliability, and flight performance subprograms.

LIQUID AND SOLID WEIGHT MODEL

The liquid and solid weight model determines the oxidizer and solid fuel weight required to match the ASRM thrust versus time profile and specific impulse (Isp) tables in response to the input values of mixture ratio, chamber pressure, and nozzle expansion ratio. Oxidizer tank ullage is assumed to be 2% of the total volume. Reserve propellant is assumed to be 2% of the propellant weight.

PRESSURE VESSEL MODEL

The pressure vessel model determines the pressurant tank volume, tank size and shape, pressurant mass initially in the pressurant tank and pressurant mass in the oxidizer tank at thrust termination. The model can use either pure helium or Tridyne (a mixture of helium, hydrogen, and oxygen) as the pressurant.

TANK AND INTERSTAGE MODEL

The tank and interstage model determines the tank wall thickness (including gas generator case thickness), ellipsoidal ratio of the dome, and the tank weight. Upper and lower dome thicknesses are determined from tank pressure and hydrostatic head pressure developed due to a 3g maximum ascent acceleration. The tank fabrication process with the IM7 carbon fiber composite material was assumed to allow tapered wall thickness based on the pressure gradient from upper to lower dome. The wall thicknesses are evaluated for local buckling and stiffeners are added, or a slight increase in wall thickness made if required. An aluminum liner is used inside the composite shell of the oxidizer tank and Tridyne tank to prevent direct contact of the fluids with the composite material. No insulation is used on the tanks. Other options evaluated, but not used in the final configurations, were aluminum oxidizer tanks, steel gas generator case, and inverted aft tank dome to shorten the tank length.

RELIABILITY MODEL

The reliability model computes the reliability of each subsystem and the reliability of the overall system. Depending on the number of required components and redundant components used in the system, each delivered component reliability is calculated and is available to be integrated into the subsystem reliability and the overall system reliability.

FLIGHT PERFORMANCE

The flight performance subprogram performs a trajectory simulation of the launch vehicle to main engine cutoff and analytically determines the OMS propellant to achieve a 150 nmi circular orbit at 28 degree inclination. The orbiter and external tank weight at lift-off was determined to be 1,840,600 pounds with 1,578,600 pounds of propellant and a delivered vacuum Isp of 452.4 seconds. No fluids were assumed lost from the launch vehicle during ascent except propellant delivered to the engines. The flight profile was a vertical ascent to a point where a gravity turn would deliver the vehicle to a perigee altitude of 50 nmi. The orbiter's OMS engines are used to circularize the orbit.

COST MODEL

The life cycle cost (LCC) model was developed using experience from launch vehicle and commercial aircraft programs. As in most parametric cost models, weight is the primary input into the costing algorithms.

The cost algorithms for the hybrid booster are comprised of several elements as illustrated in figure 3. Within the categories of hardware, support, facilities, ground support equipment, and launch operations, the cost associated with each line item is estimated separately.

Design engineering cost is estimated component by component. The cost is assumed to vary according to the equation:

$$\text{Engineering Dollars} = A*B*C*D*(wt)^{E}$$

where: A = complexity factor
B = off-the-shelf factor
C = design maturity factor
D = cost coefficient
E = cost exponent

Each component is assigned a design cost coefficient (D) and cost exponent value (E) based on historical data for a design with average complexity, no off-the-shelf characteristics, and a low design maturity. The complexity factor (A) usually varies between 0.5 and 2.0 to adjust the cost for lower or higher design complexity. The off-the-shelf factor varies between 1.0 and 0.0 to adjust the cost for some percentage of off-the-shelf characteristics. The design maturity factor usually varies between 1.0 and 0.0 to reflect the level of design maturity, such as obtained from component demonstrations (.80 factor) or tests of engineering models (.45 factor).

The Manufacturing cost is estimated in a similar manner as the Design Engineering cost. The cost equation is:

$$\text{Manufacturing Dollars} = A*B*C*D*(wt)^{E}$$

where: A = complexity factor
B = material factor
C = learning curve cumulative factor
D = cost coefficient
E = cost exponent

Each component is assigned a manufacturing cost coefficient (D) and exponent value (E) based on average manufacturing complexity, aluminum or steel material, and one unit. The complexity factor (A) adjusts the cost for lower or higher than average manufacturing complexity. The material factor accounts for the relative cost of manufacturing and raw materials. For example, carbon composite has a factor of 1.14. The learning curve cumulative factor accounts for multiple quantities of a component and the learning curve effect on cost as shown in figure 4. After calculating the manufacturing dollars, a 5% addition is made to account for the subsystem assembly effort. To account for final assembly and checkout, this 5% subsystem cost is added to the manufacturing dollars and the sum is multiplied by 15%. The support function costs are calculated based on the design and manufacturing costs as shown in figure 5.

The facilities cost is based on historical data as shown in figures 6-8. The facilities initial spares cost is computed as the sum of 2% launch & control center cost, 7% pad & site preparation cost, 2% vehicle assembly building cost.

The ground support equipment cost is based on historical data as shown in figures 9-12. The ground support equipment initial spares cost is computed as the sum of 5% launch control GSE cost, 15% pad GSE cost, 7% integration, assembly, checkout cost, and 50% mobile equipment GSE cost. The ground sector software cost is computed based on the number of lines of code for test and checkout, and lines of code for real time instrumentation, shown in figure 13.

The items comprising launch operations cost are a function of gross weight and launch weight, except that oxidizer cost is simply the cost of oxidizer loaded into the booster, shown in figure 13.

DESIGN STUDY

Figure 14 illustrates the four basic configurations. Each of the basic configurations were evaluated with LOX and H2O2 oxidizer, making a total of eight configurations. The overall vehicle diameter was set at 12 feet to be close to the ASRM diameter, and the chamber pressure was assumed to be 1,000 psia. To match the ASRM thrust profile, the maximum operating pressure occurs about 10 seconds into the burn and is approximately 1,100 psia. The nozzle area ratio was set at 15. A mixture ratio was selected to produce the highest vacuum Isp, and the ratio was held constant for the entire burn. A non-metalized fuel formulation (ARCADENE 399C) with very little hydrogen chloride in the exhaust was used. Fuel grain geometry was not optimized, but consideration was given to avoid high port velocities which could cause erosive burning. IM7 graphite composite structural material was used extensively. All configurations were expendable and used ablative nozzles and thrust chambers.

Weight allocations for thrust vector control, electronics, instrumentation, aft skirt, connecting truss, and nose cone were based on values corresponding to the current Space Shuttle Solid Rocket Booster. The gas generator configurations used gas from the main gas generator to power the turbopumps. The classic hybrid configurations used methane burned with some oxidizer to power the turbopumps and to gasify the oxidizer prior to injection into the fuel grain.

Tridyne, a mixture of helium, hydrogen, and oxygen, was used for oxidizer tank pressurization in the pressure fed configurations. It is flowed through a catalytic bed to produce a hot mixture of helium and water vapor. Tank pressurization in the pump fed configurations was accomplished using helium stored at ambient temperature to satisfy pump head requirements.

Incorporated into each design was a goal for high reliability. This goal was apportioned to each major component using historical data. The oxidizer feed system incorporated redundancy by using four turbopumps and a size which would satisfy flow requirements with one failure. A 1.6 factor of safety was used on structure to assure high reliability. The final reliability assessment determined that system reliability was about the same for the gas generator and classic hybrid concepts, 0.9985 and 0.9987, respectively.

The life cycle cost (LCC) for each configuration was estimated using a constant flight rate of one flight per month. As shown in figure 15, the lowest LCC was provided by the pump fed gas generator hybrid with LOX oxidizer, and the highest was provided by the classic hybrid with pressure fed H2O2. Figure 16 illustrates the comparison of LCC/payload weight ($/lb). This is the same trend as the comparison of LCC in figure 15, but the H2O2 pressure fed gas generator configuration, and both LOX and H2O2 pressure fed classic configuration have much higher $/lb because of their lower payload capability.

Figure 17 illustrates the gross lift-off weight (GLOW) comparison of the configurations. As shown, the configurations with LOX oxidizer are lower weight than with H2O2, and pump fed configurations are lower weight than pressure fed. The GLOW of the gas generator hybrid configurations are about the same as the corresponding classic hybrid configurations.

The selected configuration for further analysis was the pump fed gas generator hybrid with LOX oxidizer because it had the lowest LCC and the classic hybrid presented higher development risk due to the scaling uncertainties associate with the complex interactions between the oxidizer and the solid fuel grain. Figure 18 shows the detailed size and weight data computed by the hybrid design program for the selected configuration.

OPTIMIZATION STUDY

OPTIMIZATION TECHNIQUE

The optimization technique was presented in a previous JANNAF paper (Ref. 3). In summary, the ARES (Airframe Responsive Engine Selection) optimization methodology is illustrated in Figure 19. In this four step process, a Design Selector determines specific designs to be analyzed. The number of designs depends on the number of independent variables (sometimes called design variables). Figure 20 shows the savings in analysis time with the ARES technique compared to a "traditional" carpet plot optimization technique. As shown for six independent variables, for example, 49 designs must be synthesized and evaluated when using the ARES method, while over 4,000 would be required to perform the same level of analysis with a traditional approach. The time savings is substantial when one considers that approximately 30 minutes is required to completely synthesize one design with the design program. The number of levels required, seven in this example, indicates that the 49 designs are comprised of designs using seven intermediate values of the independent variables determined by the method of orthogonal Latin squares. The number of ARES cases is always the square of the number of levels.

The second step, as shown in figure 19, is to evaluate the designs with the hybrid booster design computer program. The objective is to determine values of dependent variables (sometimes called performance variables) for each design.

In the third step, a data regression is performed to fit quadratic curves to the data. The analysis includes only those terms in the equation which are mathematically determined to be significant. Each dependent variable has its own equation in terms of the independent variables.

In the fourth and final step, optimizations are performed on the quadratic curves. The program uses the method of steepest descent. Optimizations can be performed in different ways by constraining dependent variables and fixing selected independent variables. For this study, only unconstrained optimizations were performed. Since the quadratic curves approximate the dependent variables, part of the fourth step is to input the optimum independent variable value into the booster sizing and trajectory performance computer program to determine the dependent variable values for greater accuracy.

Figure 21 illustrates the four independent variables used in this optimization analysis. As indicated in figure 20, 25 designs must be evaluated with 5 different levels of independent values. As shown on the left of figure 21, each variable was assigned minimum and maximum values for the study. The 5 levels for nozzle expansion ratio, for example, were 7.0, 11.5, 16.0, 20.5, and 25.0. Similarly, the other variables are divided into 5 levels, and the Design Selector uses the method of orthogonal Latin squares to determine the 25 designs, represented by different combinations of independent variables, that must be evaluated. The hybrid booster design program was used to determine the dependent variable values, shown at the right of figure 21, for each of the 25 designs.

OPTIMIZATION RESULTS

Five different optimizations were performed and compared to the baseline design. These optimizations were:

1) Minimum life cycle cost/lbm payload
2) Maximum payload weight
3) Minimum life cycle cost
4) Minimum empty weight
5) Minimum gross lift-off weight

Figure 22 illustrates the optimum values of the four independent variables for each of these optimizations. As shown, for maximum payload weight, the diameter should be as low as possible. The horizontal crosshatched bar indicates the limit of the variable for the study. Although the optimizations can be extrapolated outside the range of the regression data base, the limits were retained for best accuracy and to prevent designs that would be unrealistic to manufacture. Minimum LCC had the greatest diameter. Mixture ratio was close to the baseline value for minimum $/lb and maximum payload weight, but it was at its maximum limit for minimum LCC and GLOW. It was slightly less than the baseline value for minimum empty weight. Chamber pressure was approximately 1800 psia for most optimizations, but it was at its lower limit for minimum empty weight, and close to the baseline value for minimum GLOW. Optimum nozzle expansion ratio followed the chamber pressure trend except that it was at its maximum value for minimum GLOW.

Figure 23 illustrates the optimum dependent variable values. Life cycle cost was close to the baseline value except that some reduction was obtained for the design that was optimized for minimum LCC. Surprisingly, the LCC for the minimum empty weight was significantly greater than the other designs. The payload capability was slightly greater than the baseline for most designs, except that it was less for the minimum empty weight and minimum GLOW designs. The $/lbm payload was lower than the baseline for the minimum $/lbm design, maximum payload design, and minimum LCC design. The results indicate that lower $/lbm can be achieved though a design with lowest LCC rather than maximum payload, although the design for true minimum $/lbm is significantly different (as shown by figure 22). GLOW is approximately the same for the designs, except that the minimum empty weight design actually had significantly higher GLOW. The booster length follows the inverse of the diameter relation with the longer boosters corresponding to the smaller diameters, and vice-versa.

CONCLUSIONS

The initial comparison of eight hybrid booster configurations, including pump and pressure fed options, LOX and H2O2 oxidizer options, and gas generator and classic hybrid concepts, showed that the pump fed, gas generator configuration with LOX oxidizer had the lowest LCC and $/lbm payload. The gas generator concept is also attractive because of its lower development risk.

Through the use of advanced structural materials and an optimized design, over 40% increase in payload capability can be achieved compared to that provided by the ASRM. The design optimization study showed that the lowest $/lbm payload is achieved with a higher chamber pressure than used in the baseline vehicle (1800 versus 1000 psia). The minimum empty weight design had the highest LCC and GLOW, indicating that studies that simply minimize empty weight in lieu of performing a LCC analysis could be in error.

The results of this analysis indicate substantial increase in payload capability, reliability, safety at relatively low cost can be achieved with hybrid propulsion. Supporting test demonstrations are required to validate the performance assumptions.

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3. L. E. Fink, B. M. Dunn, "An Automated Procedure for Integrated Simulation and Optimization of Ramjet Missile Sizing and Performance", 1979 JANNAF Propulsion Meeting, Vol. 5, pp. 209-222, August 1979.

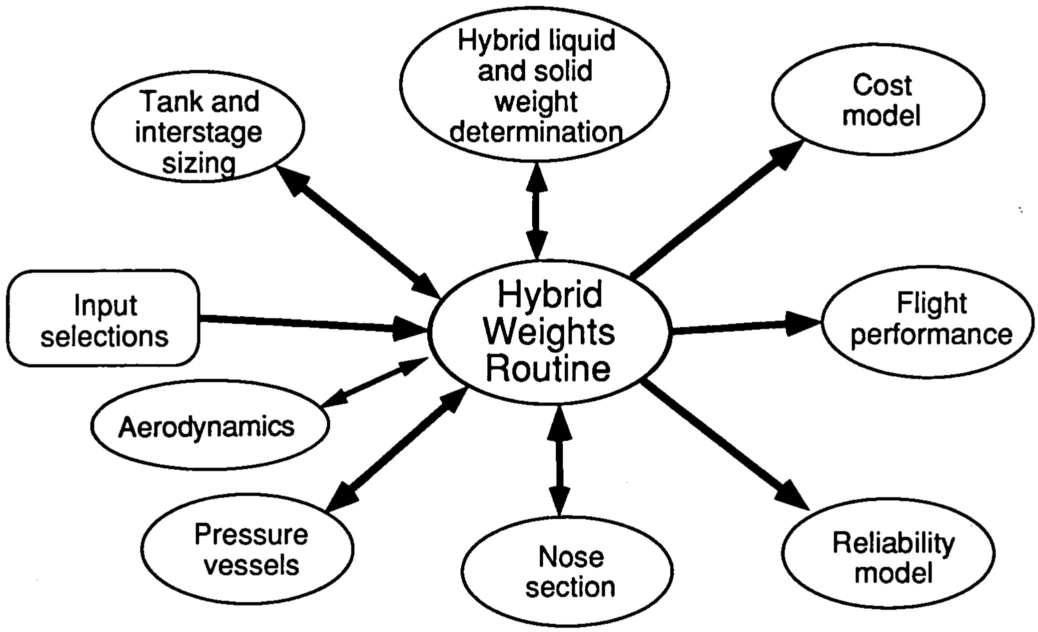

Figure 1: Hybrid Computer Model Components

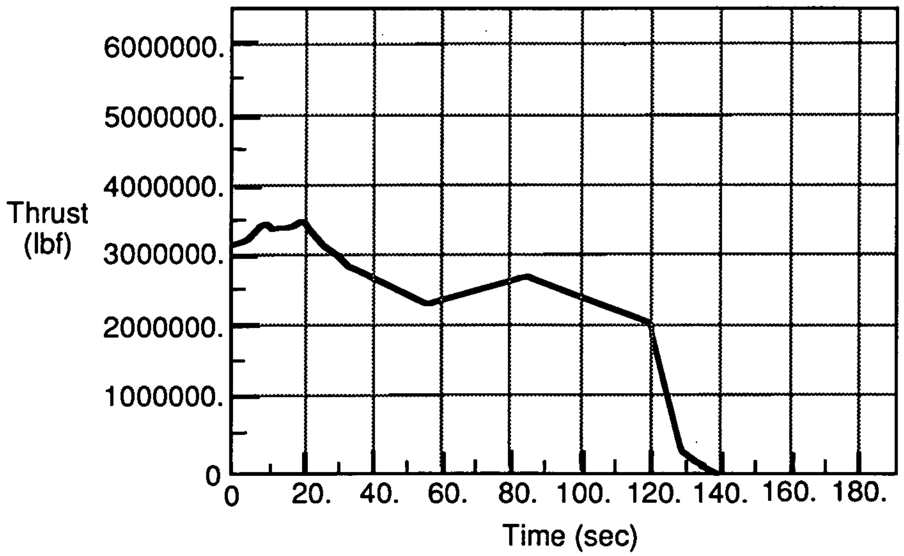

Figure 2: ASRM Thrust Versus Time

Hardware

Design engineering- component and subsystem design, drafting, development shop, testing, finance support, supervision and clerical.

Manufacturing - factory labor, quality control, subcontracts,material

Support

Systems Engineering
Software Engineering
System Test
Tooling
Miscellaneous- logistics, engineering liaison, facilities engineering, training

Facilities

Launch and control center
Pad & site preparation
Vehicle assembly & facility modification

Ground support equipment

Launch control
Pad
Integration, assembly, checkout
Mobile equipment
Initial spares
Ground sector software

Launch operations

Technical system management
Prelaunch operations checkout
Oxidizer cost (solid fuel is a hardware cost)
Mission and launch control
Recovery cost (if applicable)
Replenishment spaces

Figure 3 : Life Cycle Cost Elements

Component type	Learning curve slope
Structure	90%
Electronics	92%
Thermal	94%

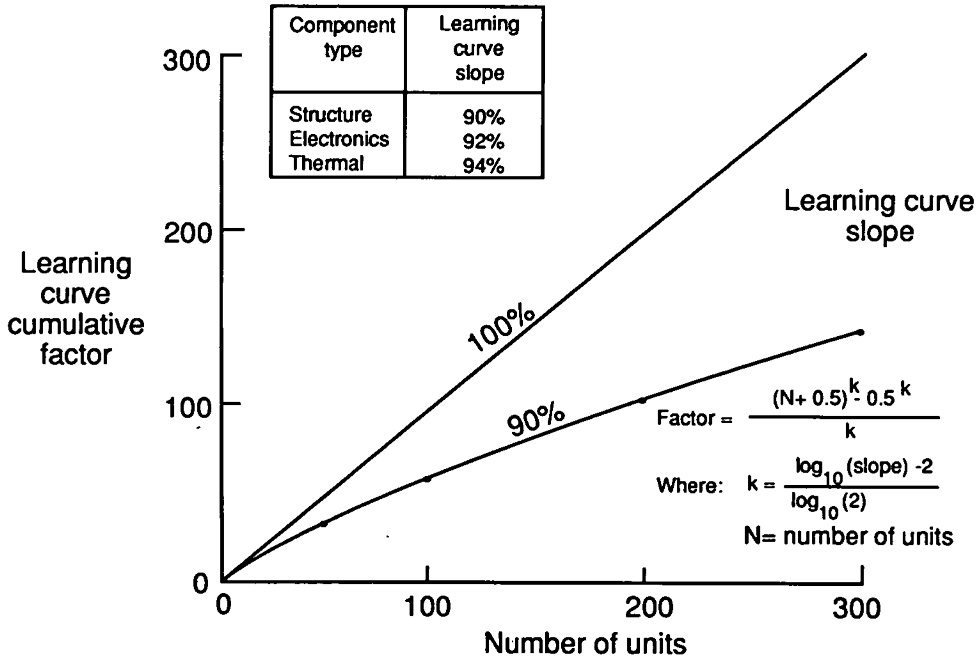

Figure 4 : Manufacturing Cost Learning Curve Factor

Systems engineering dollars are computed as :
0.323 * (Design $) ** 0.9802

Software engineering dollars are computed as :
1.370 * (Design $) ** 0.8944

Systems test dollars are computed as :
0.0006 * (Design $) ** 1.3226

Tooling costs are manufacturing dependant :
0.0045 * (Manufacturing $) ** 1.1526

Miscellaneous costs are computed as:
(0.1138 * (Design $) ** 1.0185) + (0.03 * (Manufacturing $))

Figure 5 Support Costs

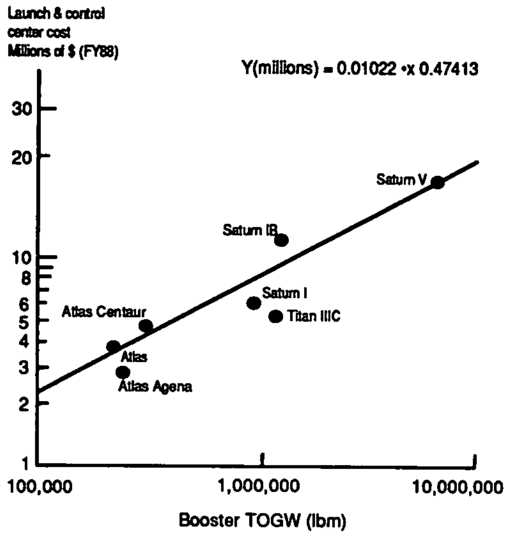

Figure 6 Launch & Control Center Facilities Cost (FCI) vs TOGW

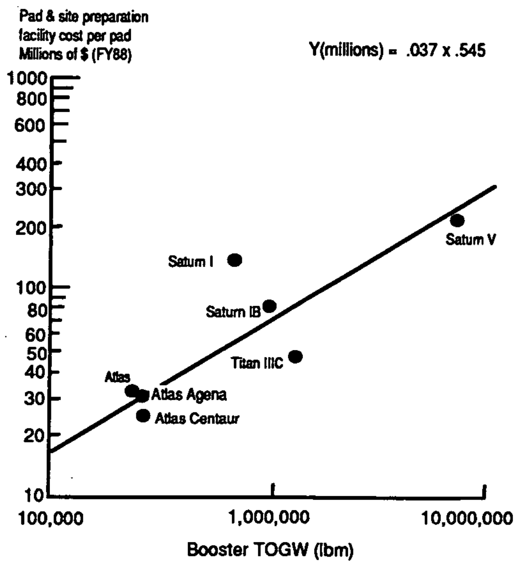

Figure 7 Pad & Site Preparation Facility Cost (FC2) vs TOGW

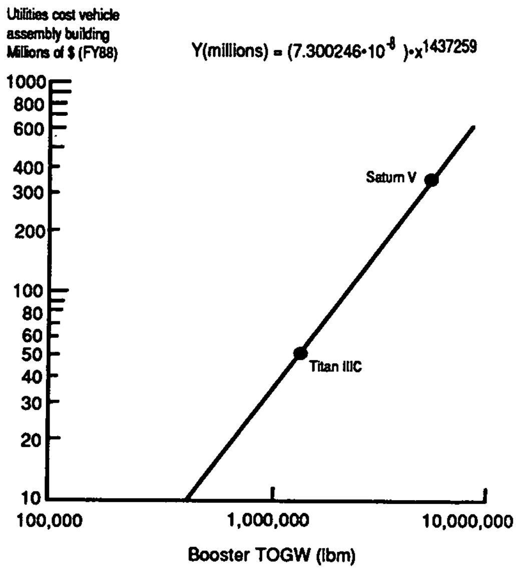

Figure 8 Vehicle Assembly Building Facilities Cost (FC3) vs TOGW

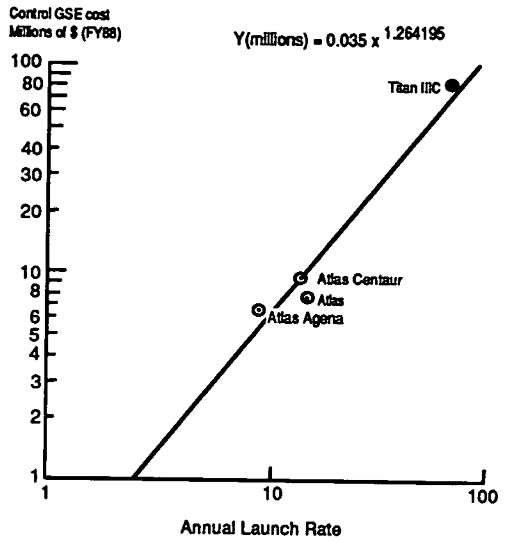

Figure 9 Launch Control GSE Cost (GSE1) VS Annual Launch Rate

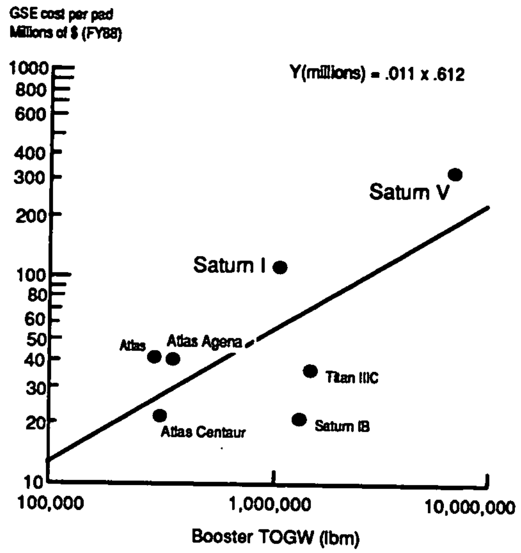

Figure 10 Pad GSE, Cost per Pad (GSE2) vs TOGW

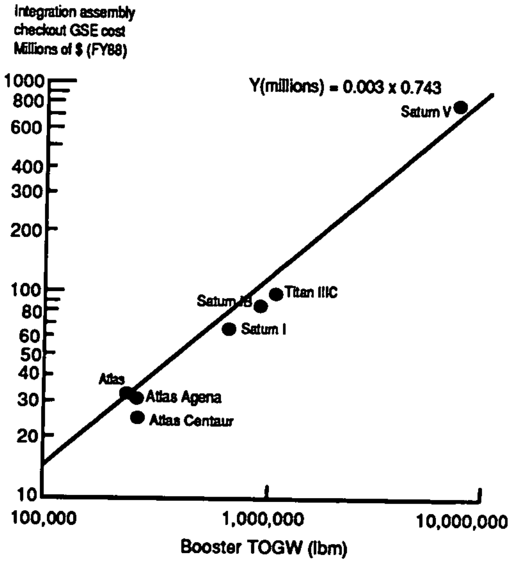

Figure 11 IACO GSE Cost (GSE3) vs TOGW

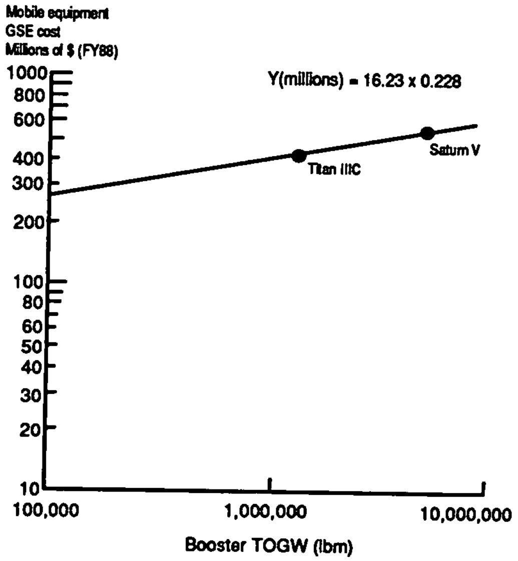

Figure 12 Mobile Equipment Cost (GSE 4) vs TOGW

Ground sector software :

$$GSE6 = 0.036 * ((KSLOC_{TST}) ** 1.12) * (K9) + 0.043 * ((KSLOC_{INST}) ** 1.20)$$

Where : K9
KSLOC = Thousands of source lines of code, real test and checkout
KSLOC = Thousands of source lines of code, real time instrumentation

The following recurring cost algorithms are for annual launch operations costs (LOC) :

Tech system management :

$$LOC\ 1 = 0.009 * (TOGW) ** 0.516) * ((L) ** 0.360)$$

Where : TOGW = Takeoff gross weight
L = Annual launch rate

Prelaunch operations checkout :

$$LOC\ 2 = 0.025 * (TOGW) ** 0.516) * ((L) ** 0.360)$$

Propellant cost :

$$LOC\ 3 = L * (W_F * C_F * B_F) + (W_O * C_O * B_O)) * 10^{-6}$$

Where : F = Fuel
O = Oxidizer
W = Propellant weight per flight (lbs)
C = Cost per lb
B = Boiloff factor
L = Annual launch rate

Note : solid propellants are included in assembly costs.

Mission & launch control : (see above)

$$LOC\ 4 = 0.010 * (TOGW ** 0.516) * ((L) ** 0.360)$$

Recovery cost :

$$LOC\ 5 = 1.77 * ((L) ** 0.534)$$

Where : L = Annual launch rate

Note : sea recovery of 1st stage booster assumed

Replenishment spares - FF/GSE:

$$LOC\ 6 = 0.10 * (FC5) + 0.20 * (GSE5) * (L** 0.05)$$

Where : FC 5 = Facilities initial spares cost
GSE 5 = Ground support equipment initial spares cost
L = Annual launch rate

Figure 13 : GSE Software and Launch Operations Cost Equations

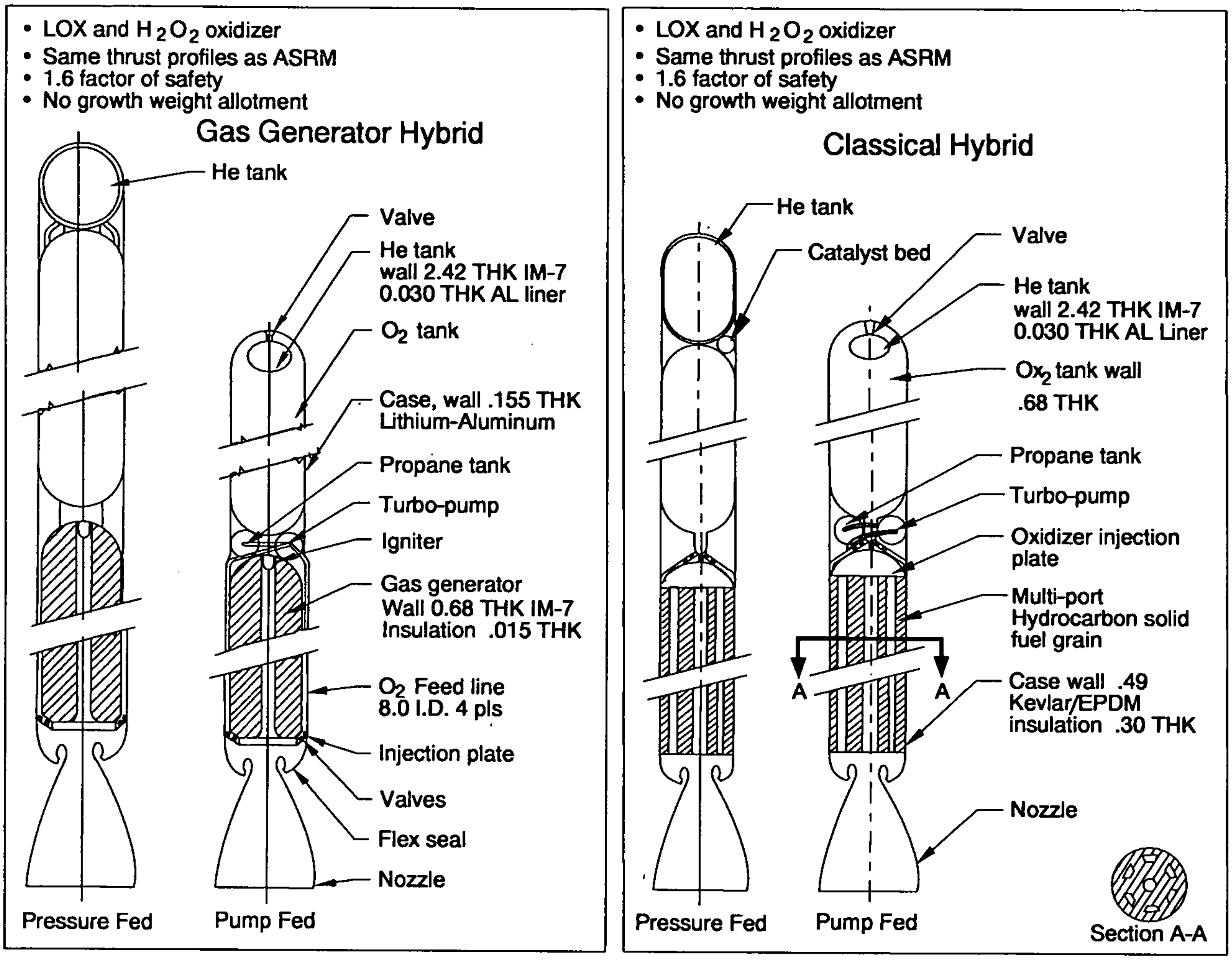

Figure 14 : Hybrid Booster Configurations

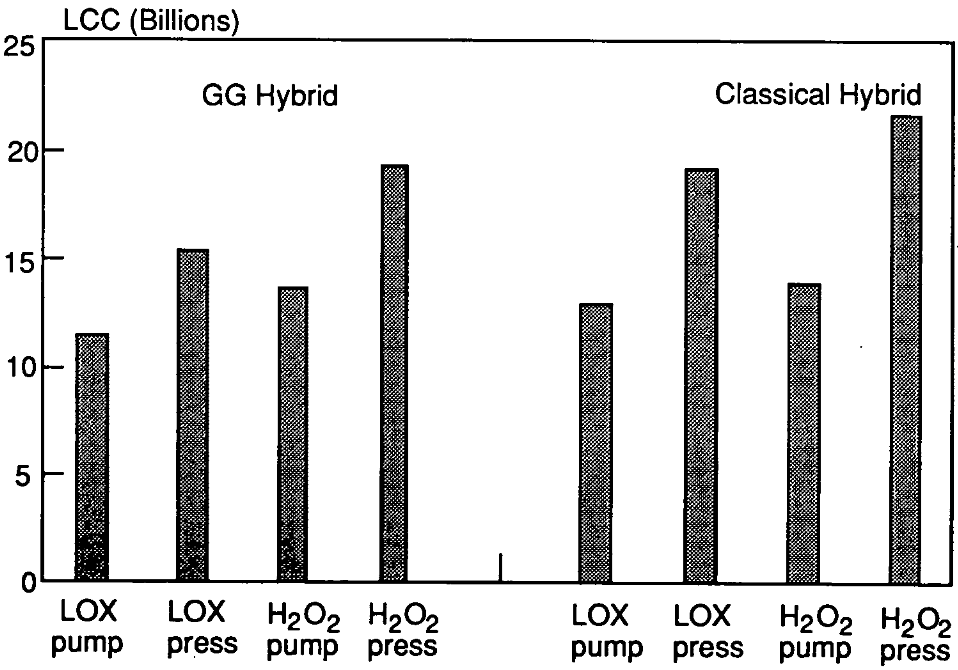

Figure 15 : Hybrid Configurations Life Cycle Costs

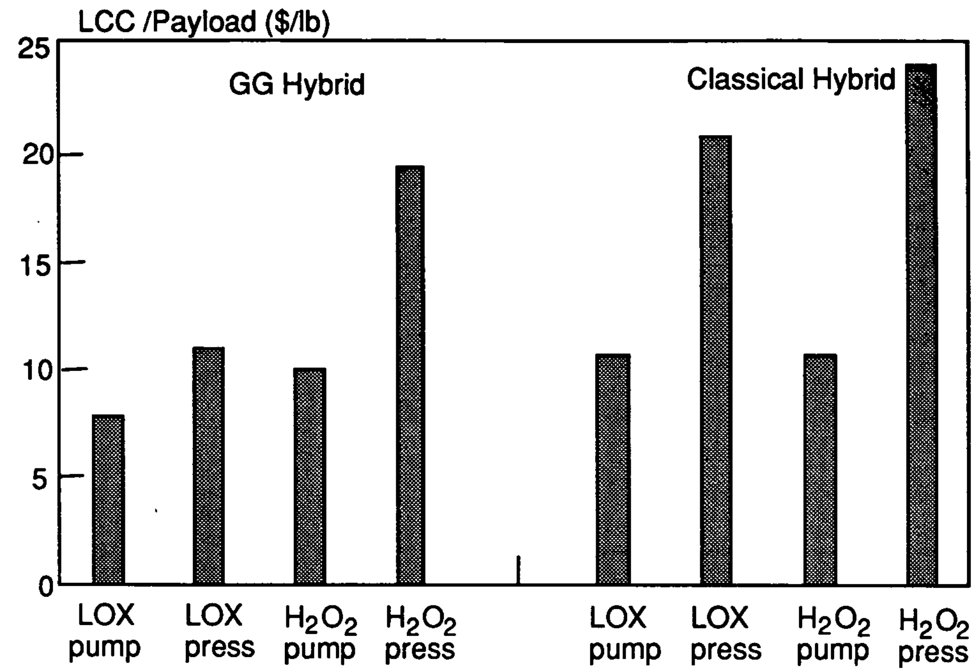

Figure 16 : Hybrid Concept Comparison, LCC/Payload

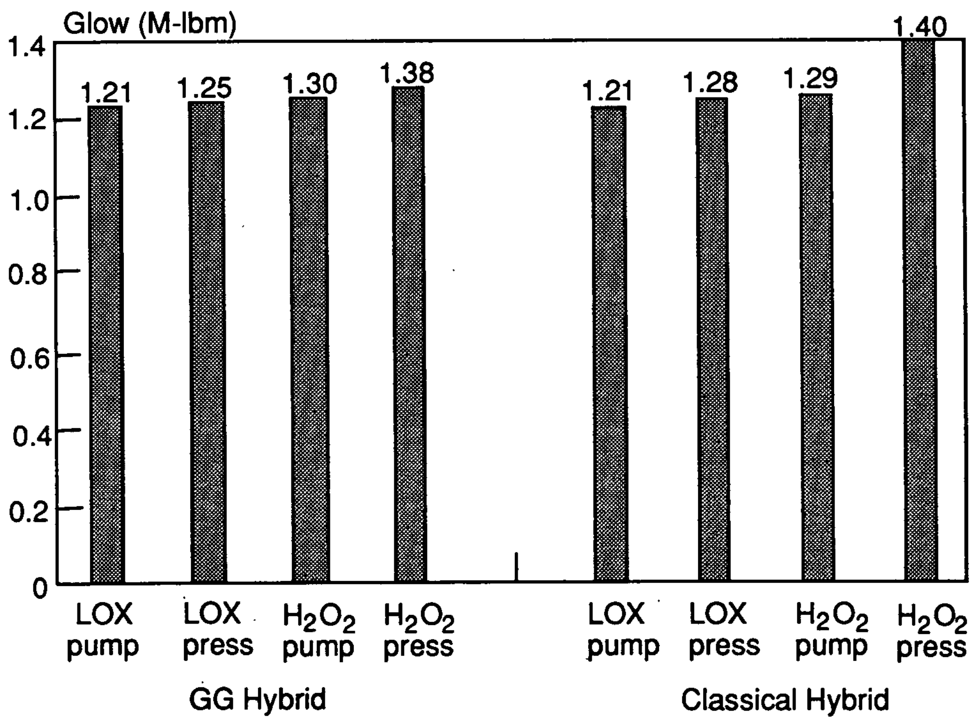

Figure 17: Hybrid Concept Comparison, Gross Liftoff Weight

Total initial weight = 1213136.14 lb
Empty weight = 83218.23 lb
Expended OX weight = 660190.69 lb
TVC OX Prop.= 0.00 lb
Turbine fuel = 5976.30 lb
Initial C.G.= 79.71 ft
Empty C.G. = 124.57 ft

Overall length = 166.64 ft
Cut off VT = 106842.77lb
Expended fuel weight = 448104.69 lb
TVC fuel Prop. = 0.00 lb
Total expended propellant = 1106295.38 lb
Cut off C.G. = 101.72 ft

Starting M.R.= 1.50
Safety factor = 1.60

Starting P.C = 1000.00 PSIA
Number of hybred units = 1

--Nose section size--

Base dia. = 14.00 ft
Nose tip Rad = 1.27 ft
C.G. from nose TIP = 10.21 ft
Location from nose TIP = 0.00 ft

Overall length = 18.91 ft
Cyl length = 0.00 ft
Weight = 1523.65 lb
To bottom = 18.91 ft

-- Helium Tank Size --

Material = IM7 carbon fiber
Outside dia.= 8.23 ft
Dome height = 2.92 ft
Dome thick = 2.869 in
Vessel weight = 3354.23 lb
Init weight = 4697.94 lb
He weight = 1176.00 lb
Init press = 1,0000. PSIA
Location from nose TIP = 13.08 ft

Length = 5.83 ft
Cyl Length = 0.00 ft
Cyl thick = 0.000 in
Aluminum liner = 44.20 lb
Shutdown weight = 3544.04 lb
C.G. from Cyl TOP = 2.91 ft
Final Pres = 144. PSIA
To bottom = 18.91 ft

-- Helium Tank Valving System--

HE Pyro value wt = 14.91 lb — Quantity = 2
Pressure regulation wt = 17.81 lb — Quantity = 1
HE service valve wt = 29.81 lb — Quantity = 1
Total value wt = 77.43 lb

-- Iner Stage (nose to OX tank)--

Material = 2219-T87 Aluminum
Dia. TOP= 14.00 ft
Length = 5.00 ft
Wall thick = 0.040 in
C.G. from TOP = 2.50 ft
Location from nose TIP 16.91 ft

Dia BOT = 14.0 ft
Weight = 137.03 lb
Stiffiners required = 0
To bottom = 23.91 ft

-- Oxidizer Tank--

Material = IM7 carbon fiber
Diameter = 14.00 ft
Dome height = 4.53 ft
Upper dome thick = 0.036 in
Cylinder thick = 0.111 in
OX tank vol. = 9639.66 ft
TOP oxidizer weight = 673593.63 lb
Residual oxidizer = 199.13 lb
Insulation = 0.00 lb
Init weight = 077312.63 lb
Init C.G. from cylinder TOP = 29.27 ft
Upper dome press = 93.PSIA
Location from nose TIP = 19.40 ft

Tank length = 70.31 ft
Cyl Len. = 61.29 ft
Lower dome thick = 0.072 in
Stiffiners required = 0.
Vessel weight = 2442.97 lb
Reserve oxidizer = 13203.81 lb
Pres gas weight = 1153.91 lb
Final C.G. = 21.16 ft
Lower dome Press = 187. PSIA
To bottom = 85.20 ft

-- LOX Valving System --

Oxidizer value WT = 241.17 lb — Quantity = 4
Oxidizer Pyro WT = 320.59 lb — Quantity = 4
Methane throttle value WT = 39.54 lb — Quantity = 4
OX relief value WT = 4.44 lb — Quantity = 1
Total value WT = 1730.23 lb — Quantity = 1

-- Boost Pump Size --

Diameter = 1.38 ft
Weight/Pump = 292.16 lb
Pumps = 4
Delta P = 54.47 PSIA
Horse Power = 417
NS = 8825
Vapor Pres = 14.34 PSIA
Location from nose TIP = 152.69 ft

Length = 1.54 ft
Total VT = 1168.66 lb
Flowrate/pump = 1628.89 lb
Speed = 2960 RPM
Pump efficiency = 77.97%
Inlet Press = 25.00 PSIA
Pump C.G. from TOP = 0.77 ft
To bottom = 154.23 ft

-- Main Pump --

Diameter = 1.56 ft
Weight/Pump = 623.94 lb
Pumps = 4
Delta P = 1034.86 PSIA
Horse Power = 7455
NS = 1995
Vapor Pres. = 14.34 PSIA
Location from nose TIP = 154.23 ft

Length = 2.27 ft
Total wt = 2495.77 lb
Flowrate/pump = 1628.89 lb/sec
Speed = 6092 RPM
Pump efficiency = 83.06%
Inlet Press = 79.47 PSIA
Pump C.G. from TOP = 1.14 ft
To bottom = 156.50 ft

-- Turbine --

Turbine flowrate = 752.01 lb/sec
ISP reduced by 0.54%
Total Pump Assem. Len = 3.81 ft

Turbine temperature = 1800 F
Fuel required = 5976.30 lb
Total Pump Assem. Weight = 3664.43 lb

--Oxidizer Propellant Line to Combustion Chamber--

Material : AISI 301 stainless
OX line dia. = 7.00 in
Number of lines = 4
Total line Wt. 1395.89 lb

Length = 68.89 ft
Weight/Line = 348.97 lb

-- Solid Fuel Case --

Material = IM7 carbon fiber
Diameter = 13.00 ft
Dome Ht. = 4.19 ft
Ratio port to throat area = 1.46
Solid cases = 1
Upper dome thick = 0.37 in
Stiffeners required = 0.
Case weight = 11112.80 lb
Reserve fuel = 8922.09 lb
Init weight = 46979.25 lb
Igniter = 500.0 lb
Init C.G. from Cyl. Top = 25.61 ft
Starting Press = 1000 PSIA
Location from nose TIP = 81.00 ft

Length = 55.41 ft
Cyl. Length = 51.22 ft
Init Port Rad = 2.33 ft
Grain Length = 51.22 ft
Cyl. Thick = 0.627 ft
Avrg. Del ISP = 293.66 Sec.
Total Fuel Weight = 455026.78 lb
Insulation = 3039.68 lb
Empty Weight = 14652.47 lb
Empty C.G. = 25.61 ft
Maximum Press = 1089 PSIA
To Bottom = 136.41 ft

-- Convergent Section --

Material = IM7 carbon fiber
Case weight = 349.51 lb
Total Wt. = 2421.621 ft
Length = 3.80 ft
Location from nose TIP = 136.41 ft

Insulation = 3229.90 lb
CG from Top = 1.52 ft
Outlet Dia. 5.47 ft
To Bottom = 140.22 ft

--C.G. Injector--

Injector Dia. = 5.4 7 ft
Weight = 2704.10 lb
Location from nose TIP = 140.89 ft

Length = 8.00 in
To Bottom = 140.89 ft

-- Combustion Chamber --

Material IM7 carbon fiber
Weight chamber = 138.16 lb
Total Wt = 2421.62 lb
Wall thick = 0.20 in
Length = 5.00 ft
Location from nose TIP =145.89 ft

Weight Ins = 2283.46 lb
CG from Top = 2.50 ft
Insulation Thick = 5.00 in
Outside Dia. = 5.47 ft
To Bottom = 145.89 ft

--Throat Size--

Throat ID diameter • 5.47 ft
Weight = 16656.26 lb
Location from nose TIP• 3.87 ft

Length = 4.40 ft
CG from Top = 26.39 ft
To Bottom = 150.28 ft

--Nozzle Size--

Dia. Nozzle Exit = 14.97 ft
Weight = 8711.62 lb
CG from Top = 8.18 ft
Location from nose Tip = 150.28 ft

Length = 16.36 ft
Exp Ratio = 150.28 ft
To Bottom = 166.64 ft

--TVC Actuator--

Weight = 2328.00 lb

-- Base Skirt Size--

Material = 2219-T87
Dia Top =13.00 ft
Length = 20.95 ft
CG from Top = 10.47 ft
Location from nose Tip = 136.41 ft

Dia Base = 13.68 ft
Weight = 13722.00 lb
To Bottom = 157.36 ft

-- Booster to Core Truss--

Truss Weight = 1165.56 lb

--Booster Separation System--

Separation System Weight = 1487.00 lb

-- Range Safety--

Range Safety Weight = 144.00 lb

Figure 18: Detailed Size and Weight Data for Selected Gas Generator Hybrid Configuration

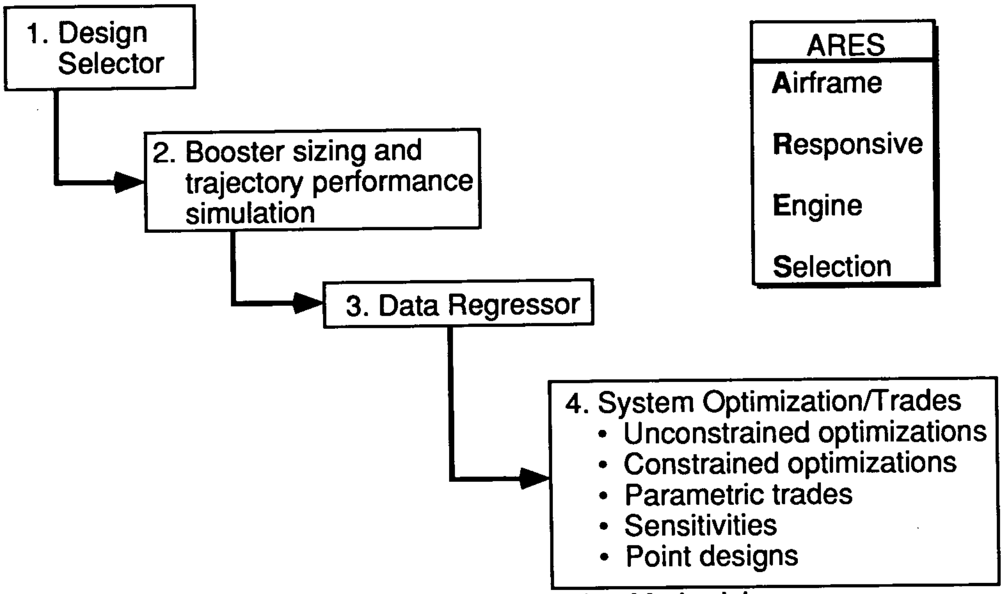

Figure 19 : ARES Optimization Methodology

Number of independent variables	Number of levels required	Required number of ARES cases	Required number of "Traditional" cases (4 levels)
N			4^N
3	4	16	64
4	5	25	256
5	7	49	1024
6	7	49	4096
7	8	64	16384
8	9	81	65536
9	11	121	262144
10	11	121	$1x10^6$

Figure 20 : Design Selector Case Definition Relationship

	Independent Variables	Min	Max
1	Mixture ratio	1.4	1.6
2	Chamber presure	700	2000
3	Body diameter	10	20
4	Nozzle expansion ratio	7	25

25 designs evaluated with 5 different levels of independent variable values

Dependent Variables
Payload
GLOW
LOX generator case weight
Total length
DDT&E cost
Acquisition cost
Non-operations cost
Operation cost
Life cycle cost
$/payload weight

Figure 21 : Optimization Variables

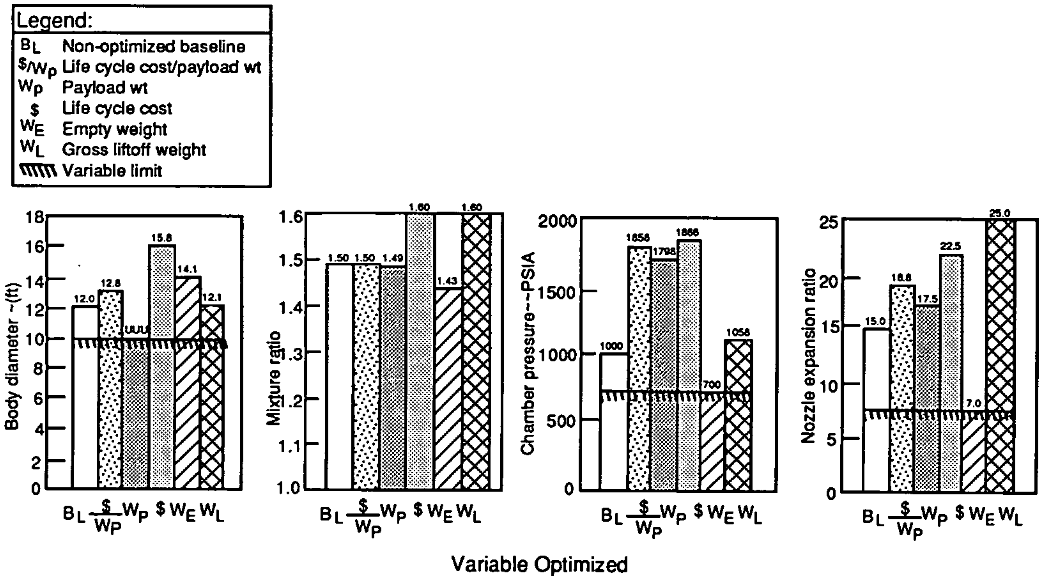

Figure 22 : Optimum Independent Variable Values

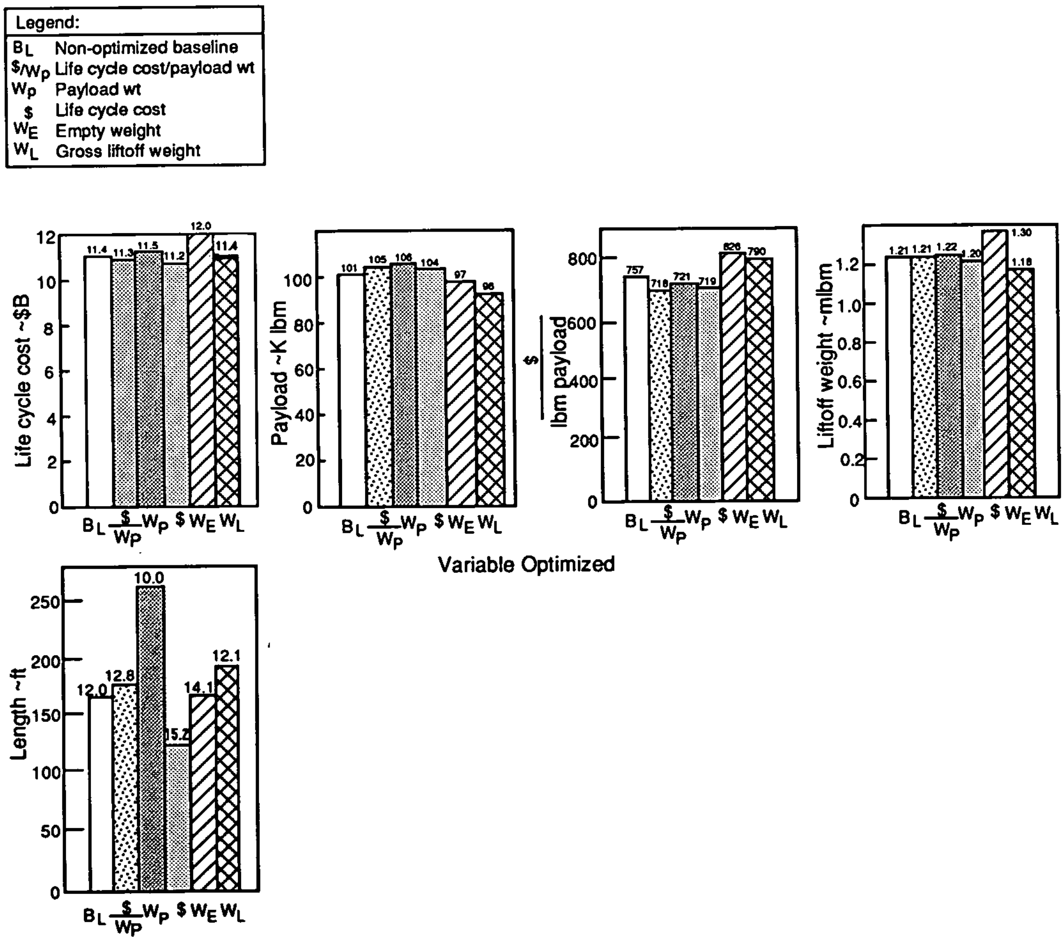

Figure 23 : Optimum Dependent Variable Values