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Booster Propulsion/Vehicle Impact Study-II Final Report

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FOREWORD

This document is prepared for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under Contract NAS8-36945, and is submitted as the technical report for the contract.

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1.0 INTRODUCTION

1.1 Preface

This report documents the study results for the Booster Propulsion Vehicle Impact Study-II which was performed from 15 July 1987 to 8 February 1988. The purpose of this study was to investigate the impact on space launch vehicle dry mass components when various propulsion options were used in the boost phase of the launch vehicle. This was done for two launch vehicles, a two stage, fully reusable vehicle and a single stage to orbit vehicle. Both vehicles are fully reusable and employ a flyback method to achieve recovery of the stages. In addition, an investigation of the design impacts on ground support and vehicle subsystems when subcooled propane is used as a fuel was made. This design impact analysis also included first order costs for ground support equipment and facilities.

This study was performed by the Space Launch Systems Company of the Martin Marietta Astronautics Group. It was conducted for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under the technical direction of Fred Braam.

The results of the study presented in this document are intended to show vehicle impacts of different propulsion options on a strictly dry mass basis. Comparisons between different options are shown on a performance basis only. There is no direct cost implications in this data other than the impact on total vehicle cost due to vehicle dry mass. The costs determined for the ground support subsystems is only for facilities and does not include manpower.

1.2 Background

1.2.1 Vehicle Impact

Current studies examining alternative rocket engine designs for use in the next generation launch vehicles primarily focus their trade studies on specific engine issues without substantial evaluation of the impact of engine design on the total launch system. This launch system impact must be determined in order to completely evaluate the merits of competing engine design issues such as: engine fuel , selection of engine coolant, usage of a translating nozzle, high and variable mixture ratio engine concepts, and improving specific impulse efficiencies. These differing designs result in different values for engine thrust, weight, mixture ratio, delivered specific impulse and fuel density, all of which affect the launch vehicle performance. A consistent analysis requires that changes in each of these parameters, resulting from a specific engine design, must be determined and then applied to a vehicle sizing procedure to quantitatively determine impacts on vehicle geometry and weights.

1.2.2 Engine Fuel Impacts

The selection of engine fuel impacts the vehicle primarily due to the resulting engine performance. However, depending upon the fuel, specific vehicle subsystems are impacted as well and may require different designs. Such subsystems as: pressurization, propellant conditioning, feed system, tankage systems are of particular interest.

In addition to impacting the launch vehicle, the selection of engine fuel may also significantly impact the ground operations facilities that support the launch vehicle. In particular, the use of subcooled propane versus normal boiling propane can require additional facilities or an increase in facility capabilities in the areas of storage, refrigeration, transfer etc.

2.0 STUDY OBJECTIVES AND SCOPE

2.1 Objectives

2.1.1 Vehicle Analyses

The primary objective of this study is to determine the design impacts on launch vehicles when various engine design concepts are used for the boost phase of the launch vehicle. The vehicles of interest are: a single-stage-to-orbit (SSTO) vehicle and a two stage fully reusable vehicle. Both of these vehicles are fully reusable by providing flyback capability for the major stage or stages. Both vehicle types are assumed to use a LOX/LH2 engine for sustainer or second stage operation.

The specific engine concepts examined in the study are: usage of three hydrocarbon fuels, RP-1, methane and propane (both subcooled and normal boiling point); using fuel as a coolant for the hydrocarbon fueled engines; using hydrogen as a coolant for the hydrocarbon engines; use of high mixture ratio LOX/LH2 engines; use of variable mixture ratio LOX/LH2 engines; and use of a translating nozzle on the boost phase engine. In addition to these concepts, we conducted analyses to find the vehicles' total dry mass sensitivity to engine thrust to weight, engine mixture ratio and engine specific impulse when using hydrocarbon fueled engines. The range for the engine specific impulse sensitivity analysis incorporated the far term performance levels; those expected levels of performance expected in the next five to ten years.

Finally, the study examined the impact on the two stage fully reusable launch vehicle of using crossfeeding of propellants from the booster to the second stage. This was done for all the hydrocarbon fueled engines using hydrogen as a coolant as well as for an all hydrogen vehicle.

2.1.2 Subcooled Propane Analyses

An additional objective for the study was to determine a first order impact on design and cost, for ground operations facilities and the launch vehicle resulting from using subcooled propane versus normal boiling point propane as a fuel. Various approaches for storing, subcooling, transferring and maintaining the subcooled state were examined. The cost was estimated based upon the most significant differences in facilities and equipment for using subcooled versus normal boiling point propane.

2.2 Scope

As defined in the statement of work, the vehicle impact is characterized as an impact on total dry mass, various subsystem dry masses and vehicle geometry. The subsystem mass of primary interest is the propulsion system mass, which for this study does not include the tankage system mass. The propulsion system mass is further broken down into engine package mass and all other propulsion subsystem masses, which consist primarily of the pressurization and feed systems. A more detailed break down can not be justified based upon the differences found during the vehicle impact analyses.

2.3 Task Flow and Schedule

The study was broken into two tasks corresponding with the two objectives. Task 1 was the performance, or vehicle, impact analyses that examined the impact on the vehicles due to use of the different engine concepts. Task 2. was the subcooled propane impact analysis. Both tasks were further broken down as illustrated in Figure 2.3-1.

Task 1 has five subtasks. Subtask 1.1 establishes the baseline configurations for both types of vehicles. These configurations, once approved by the NASA, were used as the basis of the remainder of the study. Subtask 1.2 is the generation of each reference vehicle, which uses LOX/LH2 engines for all phases of flight, the analysis of the primary engine fuel/coolant trades and the determination of vehicle sensitivities. Subtask 1.3 was a vehicle design investigation for the two stage configuration. Here, the issue was whether cross feeding propellants from the first to second stage was advantageous from a total vehicle dry mass basis. Subtask 1.4 examined the impact on the vehicle designs when high mixture or variable mixture ratio LOX/LH2 engines were used in the boost phase of flight. Subtask 1.5 examined the impact on the vehicle designs when a translating nozzle was used on the boost phase engines.

Task 2 has only three subtasks. Subtask 2.1 was the generation of various design options for ground support and vehicle equipment when subcooled and normal boiling point propane is used as a fuel in the two stage launch vehicle. Subtask 2.2 was the evaluation of the various design options and selection of the best alternatives from a technical and cost basis. Subtask 2.3 was the determination of the costs for the identified ground support and vehicle equipment.

The schedule for study completion is shown in Figure 2.3-2. Each subtask is indicated along with its planned duration. The study started on 15 July 1987 and ended on 8 February 1988. There were two scheduled reviews, one at the mid-point of the study and a final review at the end.

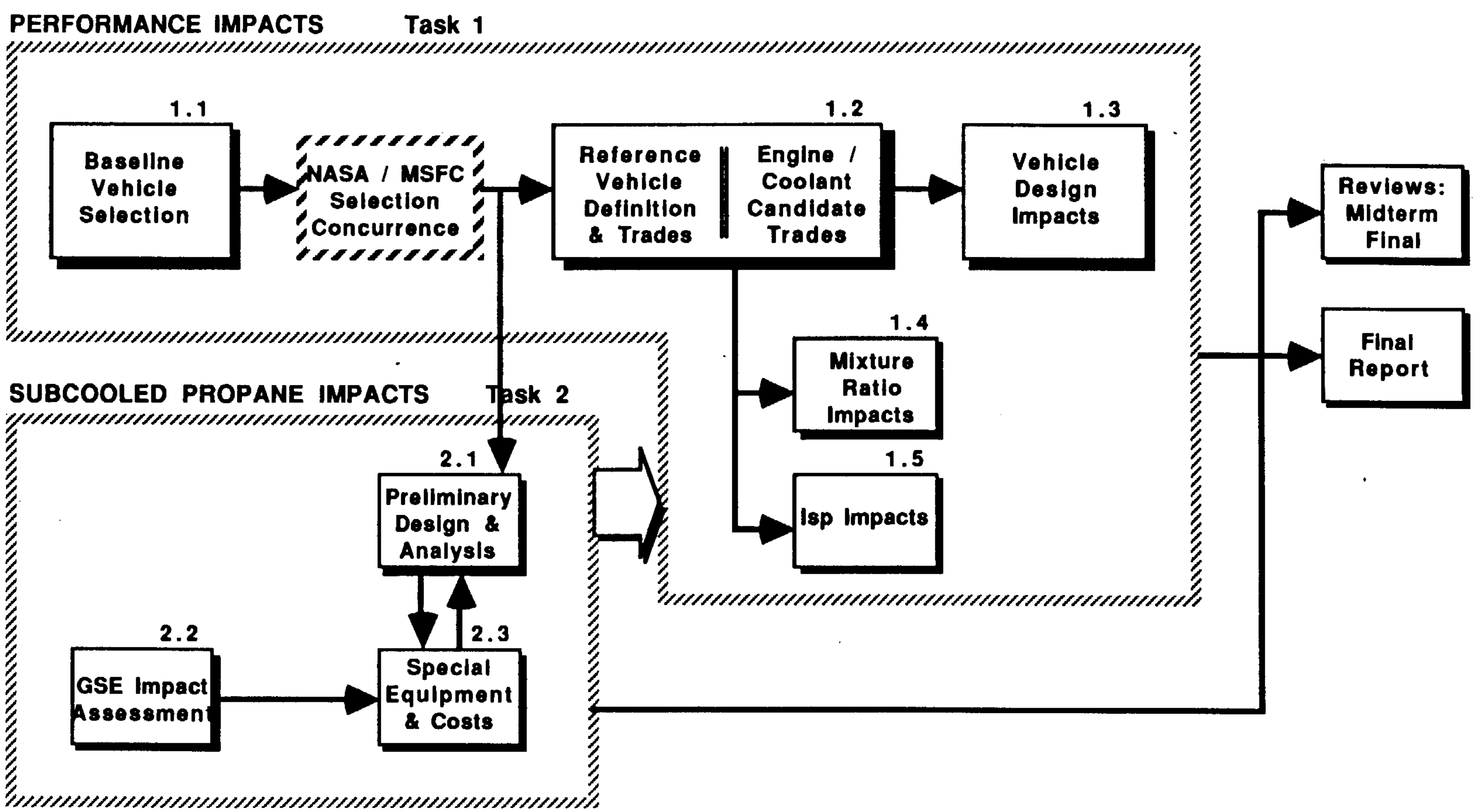

Figure 2.3-1 Task Break Down

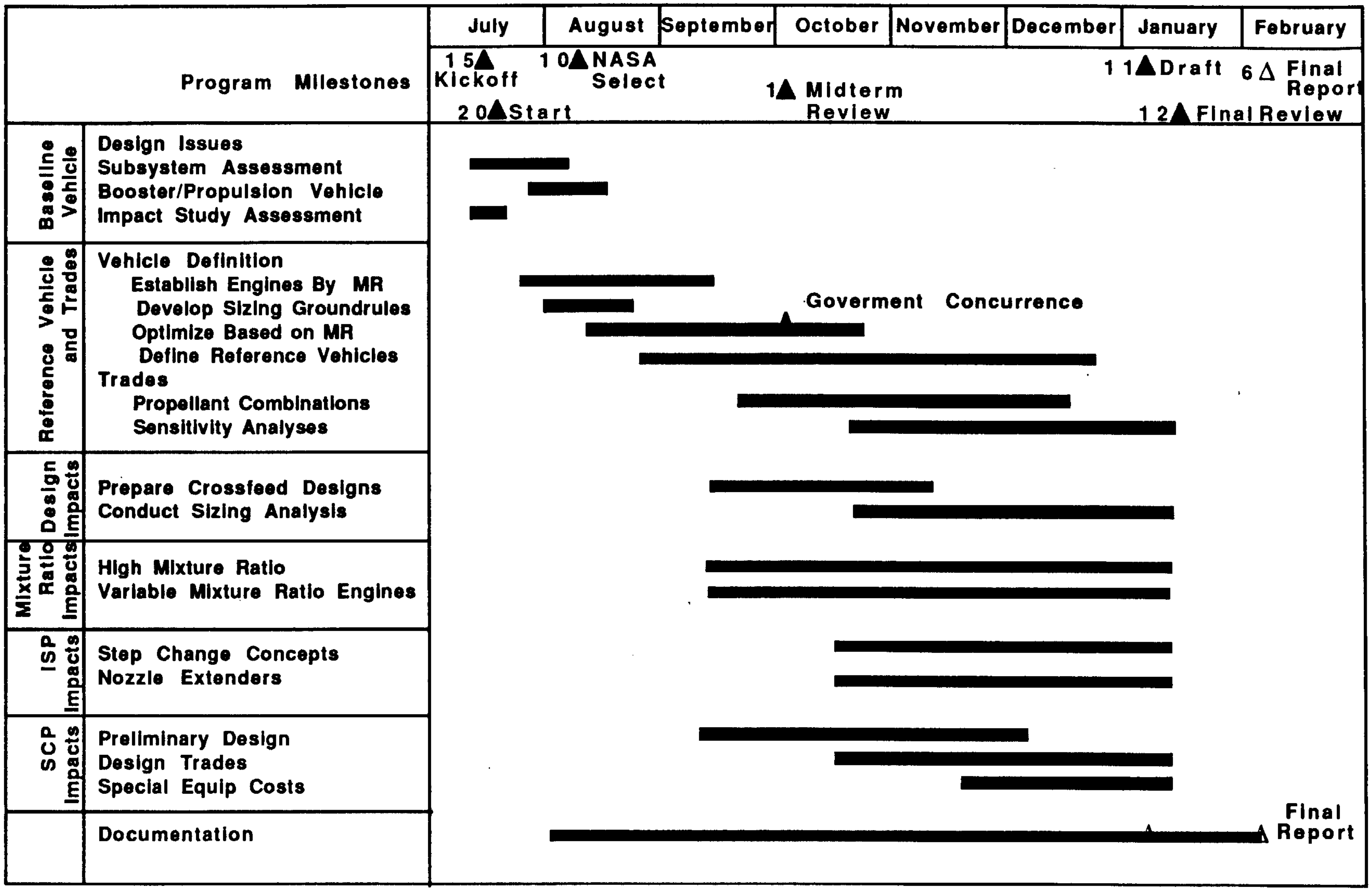

Figure 2.3-2 Study Schedule

3.0 TASK 1.0

3.1 Objective and Summary

Task 1.0 is the performance impact analysis. In this task, consisting of several subtasks, the reference vehicles are established and the impacts on the reference vehicles due to various engine options are determined. The subtasks for Task 1.0 were broken out as shown in Figure 2.3-1.

3.2 Vehicle Sizing Analysis

All of the subtasks for Task 1.0 utilize the sizing and performance models developed to conduct vehicle impact analyses. To determine the vehicle impact, in terms of weight and geometry, an integrated sizing/performance analysis is necessary to validate the predicted performance and vehicle size.

3.2.1 General Procedures

3.2.1.1 Conduct Sizing/Performance Analysis

The typical vehicle sizing/performance procedure for a single case is straightforward. First a reference vehicle design is established. This design specifies the required vehicle performance and details the system design ground rules. Once a reference vehicle has been established, the varying sizing parameters of interest are placed as input conditions to the sizing model. The sizing model then determines the revised vehicle design based upon the input conditions. A performance analysis is then conducted to validate predicted, or required, vehicle performance in terms of payload delivered to a certain orbit from a specified launch site.

3.2.1.2 Determine Parameters for Optimizing Vehicle

Depending upon the vehicle design and engine option, various sizing optimization analyses may need to be conducted in order to determine an optimum vehicle design for each engine option. Key optimization criteria for vehicle sizing are: staging velocity (or duration of boost phase) and the ratio between upper stage (or sustainer phase) total thrust to boost phase total thrust for parallel burn mode vehicles. These parameters affect the amount of energy, or propellant, that the vehicle requires to properly perform the designated mission. The propellant required largely dictates the total vehicle dry mass. Both of the mentioned parameters were used in this study for optimization of vehicle design.

3.2.1.3 Identify Sizing Trends and Establish Optimum Configuration

After optimization analysis, the sizing trends are determined in order to select an optimum vehicle configuration for the particular design option. Additional modifications to the input file may be necessary to further refine the analysis based upon the intermediate results. As each optimum vehicle design is generated for the different engine options it can be compared to the reference case, and to other design options, to establish quantitative differences in terms of weight and geometry.

3.2.2 Model Discussion

This study employs two sizing models, one for each type of vehicle, and a performance model. The SSTO sizing model is appropriately called SSTO and was obtained from J. Martin of NASA-Langley Research Center. He has used this program for numerous studies in the past. This model is tailored for a specific SSTO concept and so dictated the baseline SSTO, see discussion in section 3.3.3.1. The sizing model for the two stage vehicle is an in-house developed product called WASP (weight and sizing program). Both programs have input parameters that can be varied to represent different engine options as well as a multitude of other vehicle design parameters such burn mode, performance required, materials used in structure, fuel types etc. The performance model is called FLYIT. A more complete description of each model follows.

3.2.2.1 SSTO

Modifications from Original

The SSTO program was slightly modified from the original obtained from J. Martin. The modifications included: adaptation to allow calculation of vehicle design while using a whole number of engines, modified performance program interface in order to work with FLYIT. Finally, some slight alterations were necessary to allow the program to work in a PC environment rather than the original mainframe computer environment. None of these changes materially affected the algorithms, or the calculations made, in the program.

General Flow of Analysis Using SSTO

The basic flow of the analysis, and the use of the SSTO program, is illustrated in Figure 3.2-1. The performance model determines a target mass ratio for the SSTO given the input performance conditions, which includes engine performance characteristics. The target mass ratio is based upon an initial guess for total vehicle weight and the calculated burnout weight determined by the performance program. The SSTO program then proceeds to estimate subsystem dry weights and propellant weights iterating until the target mass ratio is achieved. In addition, the vehicle geometry is also calculated. The resulting weights generate a new total vehicle weight. If this weight is sufficiently different from the initial guess, then the program generates an input file for the performance model in order to determine a new burnout mass and mass ratio. This process is repeated until predicted performance and the vehicle size are consistent. The final vehicle results are stored for later evaluation. Multiple cases can be consecutively processed in a similar fashion in batch mode.

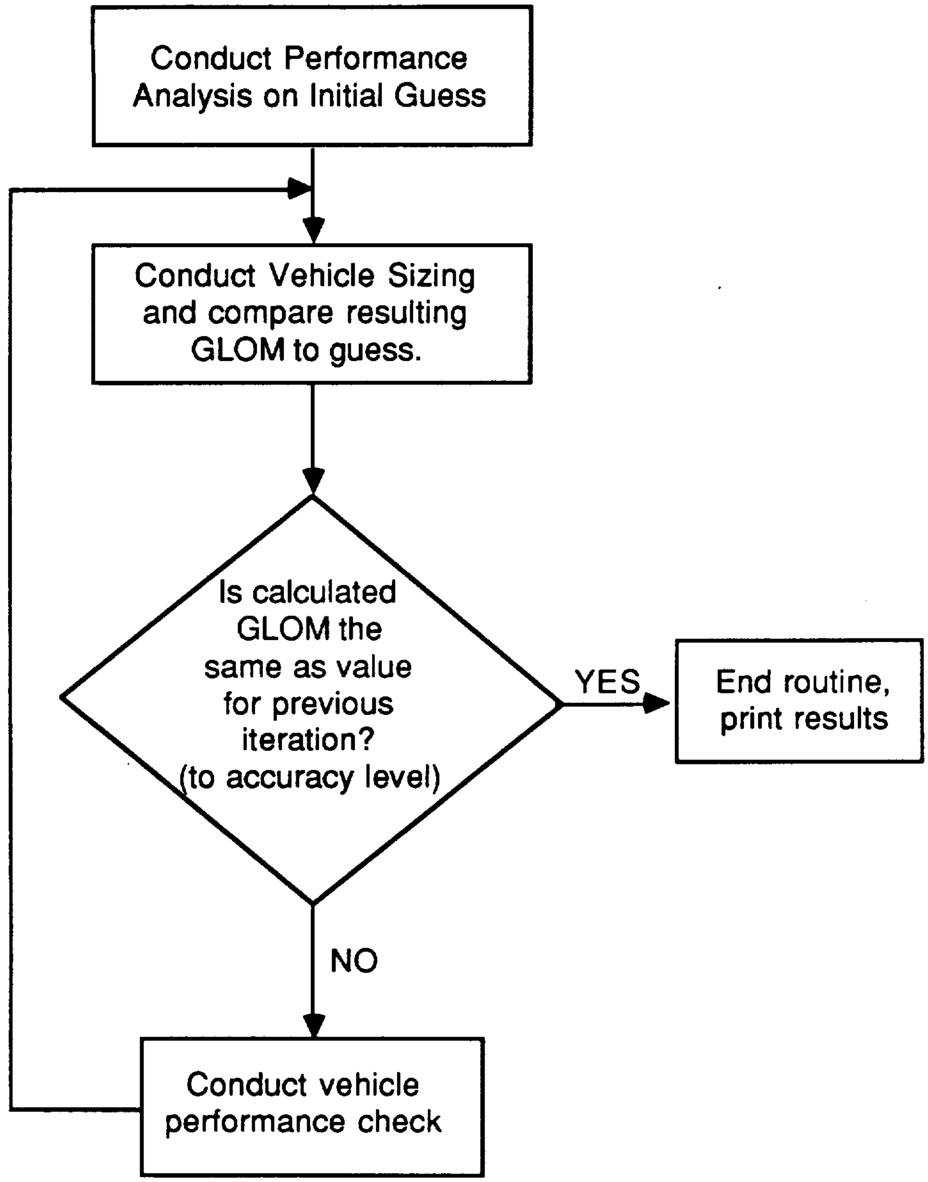

Figure 3.2-1 Analysis Flow for SSTO

Detailed Description of SSTO Model

After reading the input conditions, SSTO determines the engine thrust requirements for an initial guess on total vehicle weight. The thrust requirements are determined for two thrust phases during the total mission. The first phase is the boost phase and may have two different engine types operating during this phase. The second phase is called the sustainer phase and typically only one engine type is assumed to be operating. The thrust estimates for each burn phase then determine either the number of engines or the thrust level required by each type of engine based upon the input option selected. The number of engines and thrust levels dictate the required engine weights.

The program then determines the propellant weights for each burn phase and the required volumes for these weights. Vehicle geometries are then determined based upon propellant volumes and the baseline, built-in configuration assumptions. Subsequently, other subsystem dry weights are calculated. This process continues until the required mass ratio is satisfied. The resulting total vehicle weight is then compared to the input guess to determine whether additional performance evaluation is required.

Typical Output

Typical output of the program is shown in Figure 3.2-2 for the reference SSTO vehicle, see section 3.4.4.1. This output is based upon the sizing assumptions and ground rules incorporated into the input files to the program.

3.2.2.2 WASP

Configuration Variant

For the investigation of the two stage vehicle a sizing program was used that is a variant of other programs developed at Martin Marietta Astronautics Group. The general program is called WASP and is a FORTRAN program that runs on a personal computer. This program calculates total vehicle and subsystem weights and vehicle dimensions for a wide variety of input conditions. The program version used for this study was tailored to the specific geometry of the selected baseline configuration.

General Flow

The general flow of the two stage vehicle analysis using the WASP program and its iteration with the performance model is illustrated in Figure 3.2-3. Using the input file that contains the sizing assumptions for a specific case, the WASP program iterates the amount of propellant required by each stage, and the subsystem dry weights consistent with the propellant weights, until the vehicle satisfies the input ideal delta velocity requirements. The ideal delta velocity is the sum of the required orbital velocity and "velocity losses". The program generates an input file for the performance model FLYIT based upon the calculated vehicle design. FLYIT determines the performance of the vehicle. If the vehicle is undersized for the actual mission, (i.e. the velocity losses are greater than expected) then the vehicle will run out of propellant before reaching the required orbital speed. This "velocity error" is added, via a batch file process, to the velocity loss term in the WASP input file for the next iteration of vehicle sizing. This process continues until vehicle size and performance is consistent. The vehicle is considered "properly sized" when it burns out within .3048 mps of the required burnout speed.

Detailed Description

The WASP program first reads the input file which contains the vehicle design parameters, including the engine design characteristics such as specific impulse, thrust to weight, mixture ratio etc. Using initial estimates of stage propellant weights, the program begins the major sizing loop.

The sizing loop begins by using the propellant weights and the previous iteration's estimate of stage dry weights combined with input acceleration requirements and vehicle thrust ratio, the ratio of second stage total thrust to first stage total thrust which is used as a vehicle design parameter for parallel burn vehicles, to determine stage thrusts. These thrusts are determined for three phases of stage/vehicle flight: stage ignition, stage burnout and total vehicle thrust at maximum dynamic pressure. The calculated thrusts and known weights determine the vehicle accelerations during the three phases.

With the accelerations and required internal tank pressures, the program determines combined axial and bending loads on the major structures of the stages and the dynamic pressures inside the propellant tanks during the three phases described above. The maximum loads and pressures are used to determine the structure sizes, and subsequently the structural weights, using input material properties and simplified structural strength algorithms. Other subsystem weights are calculated using a wide variety of weight estimating relationships.

The loop finishes by summing subsystem weights and propellant weights to determine total stage weights. The stage weights and engine performances are then used to determine ideal velocities generated by each stage. The calculated ideal velocities are compared to the required ideal velocities. If the absolute difference between the two is larger than the accepted error level (generally less than .1 fps) then the program re-estimates the stage propellant required and begins another iteration.

After the program generates a vehicle that satisfies the required performance in terms of ideal velocity, an input file for the FLYIT model is created so that performance can be validated. Once sizing and performance are consistent, WASP generates an output file that contains all the vehicle design data. Pages one and two of the output for the reference vehicle, see Section 3.4.4.2, is shown in Figures 3.2-4 and 3.2-5 respectively.

```
                          * M A S S   R E P O R T *
                          *************************
          CASE  81    TRAJECTORY BURNOUT MASS=     144524.20  kg
       BOEING SINGLE-STAGE-TO-ORBIT CONCEPT  -- 13.6 METRIC TON PAYLOAD
 1.0 WING GROUP                                                 8342.   kg
   0.TAIL GROUP                                                 1802.   kg
   0 BODY GROUP                                                28475.   kg
       BASIC STRUCTURE                          9352.   kg
       THRUST STRUCTURE                         3440.   kg
       RP-1 TANK                                   0.   kg
       LOX TANK                                 6490.   kg
       LH2 TANK                                 8565.   kg
       BODY FLAP                                 629.   kg
 4.0 INDUCED ENVIRONMENT                                       13138.   kg
 5.0 LANDING GEAR                                               3931.   kg
 6.0 PROPULSION                                                28821.   kg
 7.0 PROPULSION, RCS                                            1312.   kg
 8.0 PROPULSION, OMS                                            1455.   kg
 9.0 PRIME POWER                                                1428.   kg
10.0 ELEC CONV AND DISTR                                        1957.   kg
11.0 HYDRAULICS AND SURFACE CONTROLS                            6019.   kg
13.0 AVIONICS                                                   2248.   kg
14.0 ENVIRONMENTAL CONTROL                                      1989.   kg
15.0 PERSONNEL PROVISIONS                                        763.   kg
16.0 MARGIN                                                     7286.   kg
    DRY WEIGHT                                                108965.   kg  ( .765

17.0 PERSONNEL                                                  1290.   kg
19.0 RESIDUAL FLUIDS                                            5819.   kg
    LANDED WEIGHT W/O CARGO                                   116074.   kg  ( .766

   0 CARGO (RETURNED)                                          13600.   kg
    LANDED WEIGHT                                             129674.   kg  ( .747
    ENTRY WEIGHT                                              129674.   kg  ( .747
23.0 ACPS PROPELLANT                                           11334.   kg
                 RCS                   2684.  kg
                 OMS                   8649.  kg
24.0 CARGO DELIVERED                                               0.   kg
25.0 ASCENT RESERVES                                            2686.   kg
26.0 INFLIGHT LOSSES                                             868.   kg
27.0 ASCENT PROPELLANT                                        895249.   kg
       HC                  0.   kg LOX ENG B          0.   kg  HC %=   .0
       LH2             99471.   kg LOX ENG A     795778.   kg
       HC ENGINES             .0    H2 ENGINES      6.7
       HC THRUST PER ENGINE kN 2224.1
       H2 THRUST PER ENGINE kN 2224.1

     GROSS LIFT OFF MASS                                      1039810.   kg  ( .103
```

Figure 3.2-2 Output from SSTO Program for Reference (LOX/LH2) SSTO

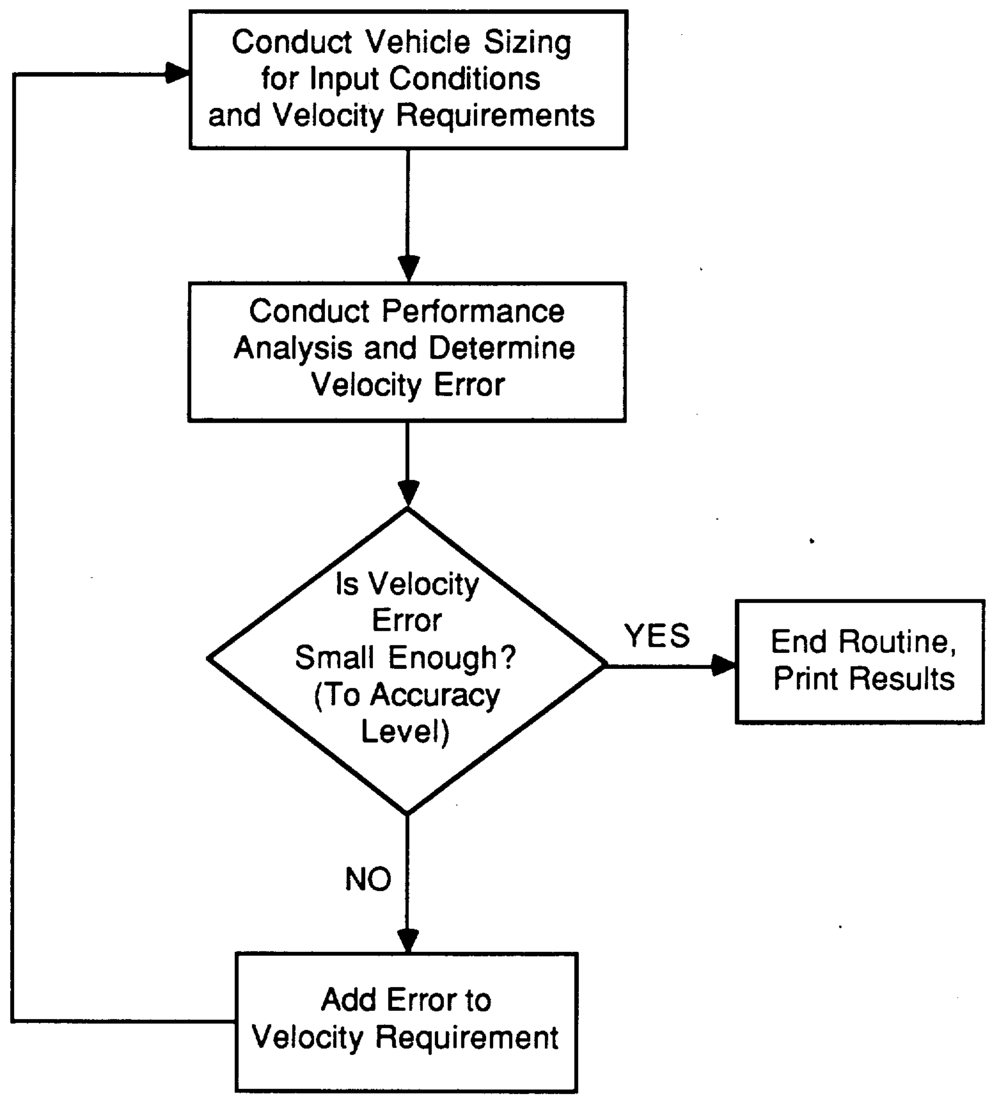

Figure 3.2-3 Analysis Flow for UFRCV

PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 34

PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3367520.
THEORETICAL VELOCITY	30147.
ACTUAL VELOCITY	24551.
VELOCITY LOSSES	5596.

	BOOSTERS	ORBITER
DRY WEIGHT	331368.	200828.
RESIDUAL WEIGHT	57545.	19018.
BURNOUT WEIGHT	388913.	219846.
TOTAL DRY WEIGHT	532197.	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		416763.
PROPELLANT WEIGHT	1779321.	914440.
WEIGHT AT LIFTOFF	2168234.	1134286.
MASS FRACTION	.8206	.8062
MASS RATIO	2.87	2.75
VELOCITY THEO	15074.	15074.
SPECIFIC IMPULSE (VAC)	439.9	463.6
(S.L.)	392.6	375.0
(STAGE 1 AVERAGE)	443.7	
THRUST (VAC)	4093963.	902004.
(S.L.)	3648094.	729619.
AXIAL ACCELERATION AT START	1.30	1.15
BURN TIME	192.	256.
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	1.00	.65
NUMBER OF BOOSTERS	2.	

Figure 3.2-4 Page 1 of WASP Output for Reference (LOX/LH2) UFRCV

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	28272.	46071.
NOSE CONE	696.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	8524.	
UPPER DOME	637.	
BARREL	6168.	
LOWER DOME	1135.	
BAFFLES	584.	
INTERTANK	3470.	
SKIN	1584.	
STRINGERS	1104.	
FRAMES/BEAMS	781.	
AFT TANK	8504.	
UPPER DOME,	410.	
BARREL	7276.	
LOWER DOME	769.	
BAFFLES	50.	
TAIL SKIRT	7078.	
SKIN	2950.	
STRINGERS	2674.	
FRAMES	1455.	
THRUST STRUCTURE	0.	
AERO SURFACES		25021.
BODY		21049.
THERMAL PROTECTION SYSTEM	1417.	38565.
SEPARATION	4113.	
RECOVERY	55348.	
LANDING GEAR		10041.
PROPULSION SYS	36004.	58430.
POWER SYSTEMS	4461.	
AVIONICS	2263.	
ACS WEIGHT	6105.	
ELECTRICAL	70.	5320.
I/F ATTACH	1201.	
CONTROLS		7230.
RANGE SAFETY	150.	1700.
GROWTH	27614.	33471.
INERT WEIGHT	165684.	200828.

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Figure 3.2-5 Page 2 of WASP Output for Reference (LOX/LH2) UFRCV

3.2.2.3 FLYIT

As previously described, vehicle sizing is performed by the WASP and SSTO models. In order to provide increased realism and accuracy in the sizing process, the trajectory program, FLYIT, is used to test a vehicle's performance. This program is written in PASCAL and runs on a personal computer. A batch file process controls the interaction between the sizing models and FLYIT. The sizing models generate relevant input files for FLYIT whereupon the vehicle trajectory is simulated from a specified launch site to the desired altitude and flight path angle.

Brief Description

FLYIT is a three degree-of-freedom flight simulator. It uses two pitch rates to target to two burnout conditions: altitude and flight path angle. The first pitch rate occurs from 10 to 20 seconds after ignition and the second acts from the point the vehicle passes 100,000 ft until it reaches final burnout. Inclination is targeted using launch azimuth. The ascent is performed in or parallel to a plane defined by the initial launch radius vector and the launch azimuth. All equations are 3-D vector equations for computing the vehicles state. The model utilizes an oblate, rotating earth model, fourth order logarithmic curve fit atmosphere, and integrates instantaneous engine pressure losses and drag during the ascent.

Limitations and Built-in Assumptions

FLYIT does not optimize the performance of a vehicle in any way. Typically a trajectory is optimized by finding the best pitch and yaw profile throughout the flight. Since FLYIT does not do this, the vehicles sized are slightly more capable than indicated. Typically this performance difference is less than five percent in delivered payload weight. In some cases (high stage-2 acceleration and/or high burnout altitude) the performance disparity becomes more significant. This occurs when the vehicle has a large (> 10 degree) nose down attitude at burnout, which wastes propellant. This was monitored in the study and corrective actions were taken when it occurred. These corrective actions consisted of upper stage engine throttling or changing the insertion point for elliptic orbits.

FLYIT, given data from a sizing model, accurately accounts for variations in all of the following parameters for each stage: Thrust-to-weight ratios, Isp's, dry weight and propellent weight (weight ratio), vehicle diameters for drag estimation, and engine exit area (expansion ratio) for pressure losses. Any vehicle utilizing a detachable payload faring, cross-fed propellants, and/or parallel stage burning are also appropriately modeled in FLYIT.

Validity

FLYIT has been validated over the past year against both POST and 3-DOpt (Martin Marietta-Michoud) for dozens of different vehicles and cases

(Shuttle, SDV, Titan-2,3,4,5, Atlas, Delta, ALS and others). In all cases performance can be matched within five percent of a fully optimized trajectory if stage-2 thrust-to-weight is used as an optimizing factor. FLYIT has also been matched against a specific ALS POST run on a second-by-second basis from launch through 100,000 ft, to verify proper integration, atmospheric modeling, and pitch control accuracy. All flight parameters (mach, altitude, velocity, etc.) matched within one percent throughout the atmospheric ascent. These results lead to a high degree of confidence that the algorithms and assumptions utilized in FLYIT are sound and appropriate for the Booster Vehicle Impact Study.

3.3 Task 1.1- Establish Baseline Vehicles

3.3.1 Objective and Summary

The objectives of this task were to define the two baseline configurations to be used in the study, to establish the ground rules for vehicle sizing and performance determination and establish the engine data to be used for the impact analyses.

An SSTO vehicle and a two stage fully reusable, unmanned vehicle were fully defined and the sizing parameters consistent with the subsystem designs for the vehicles were identified. Other performance and sizing ground rules were generated based upon interaction with the customer. These ground rules are discussed in more detail in the following sections.

Engine data for LOX/LH2 engines were obtained from the customer and selection of the engines used were based upon the ground rules. Engine data for the hydrocarbon engines, both fuel and hydrogen cooled, was obtained from the final report on the Hydrocarbon Rocket Engine Study prepared by AeroJet TechSystems[1].

3.3.2 Ground Rules, Assumptions and Inputs

3.3.2.1 Vehicle Selection

Selection of the specific configurations for the vehicles used in the study were based upon guidelines provided in the original statement of work. The required configurations were: (a) a single stage to orbit system, fully reusable with a performance of between 13,000 and 23,000 kgs to low earth orbit from ETR and (b) a two-stage, fully reusable unmanned system capable of delivering 68,000 kgs to low earth orbit from ETR. The actual vehicle designs selected had to be sensitive to the issues to be examined in the study. In order to provided a basis of comparison to previous studies examination of many vehicle designs was made in order to identify possible candidates. A wide variety of previous studies, conducted by Martin Marietta Astronautics Group and others, were evaluated for possible vehicle candidates.

3.3.2.2 Establishing Sizing Ground Rules etc.

Other than the vehicle performance requirements, there were no specified restrictions on vehicle sizing ground rules or assumptions.

3.3.2.3 Selection of Engine Data

The customer supplied the LOX/LH2 engine data to be used in the study. This data consisted of tables of parametric performance characteristics for staged combustion LOX/LH2 engines over a range of mixture ratios, thrust levels, chamber pressures and expansion ratios. Figure 3.3-1 illustrates the power cycle for the LOX/LH2 engines. Figure 3.3-2 is a typical page of the parametric data.

By contract direction the hydrocarbon engine data was to be obtained from the final reports for the Hydrocarbon Rocket Engine Study prepared by the three major rocket engine contractors: AeroJet, Rocketdyne and Pratt & Whitney[1-3].

3.3.3 Discussion of Procedures

3.3.3.1 Vehicle Selection

The final selection of the SSTO vehicle was largely dictated by the available sizing models for such a configuration. In examining previous studies it was found that a large amount of analysis of using different propulsion options in an SSTO were conducted by J. Martin of NASA-Langley Research Center[4-7]. In contacting Mr. Martin, he offered us the use of his sizing model. It was felt that the use of this model would provide results directly comparable to Mr. Martin's earlier work. However, his work, and the model , were dependant upon the advanced SSTO design generated by the Boeing Aerospace Co. as reported in NASA-CR-3266[8]. Thus, the SSTO design was dictated by the desire to use this particular model. Fortunately, this design satisfied all the study requirements.

The two stage vehicle configuration was selected from an internal data base of vehicle designs on the basis of the work already done on the configuration selected and on the manned version, which is the Shuttle II vehicle examined for the Space Transportation Architecture Study[9]. Based upon these previous investigations, it was determined that a payload capacity of 29,478 kgs was more appropriate in lieu of the 68,000 kgs identified in the statement of work; the smaller value was used for this study It should be noted that by combining the booster of the selected two stage configuration with an expendable second stage, a payload of 68,000 kgs to low earth orbit is easily achieved.

In the case of the two stage vehicle, it was also necessary to determine the burn mode and the pressurization system to be used. Preliminary sizing studies were conducted for this configuration using different fuels in the booster for the

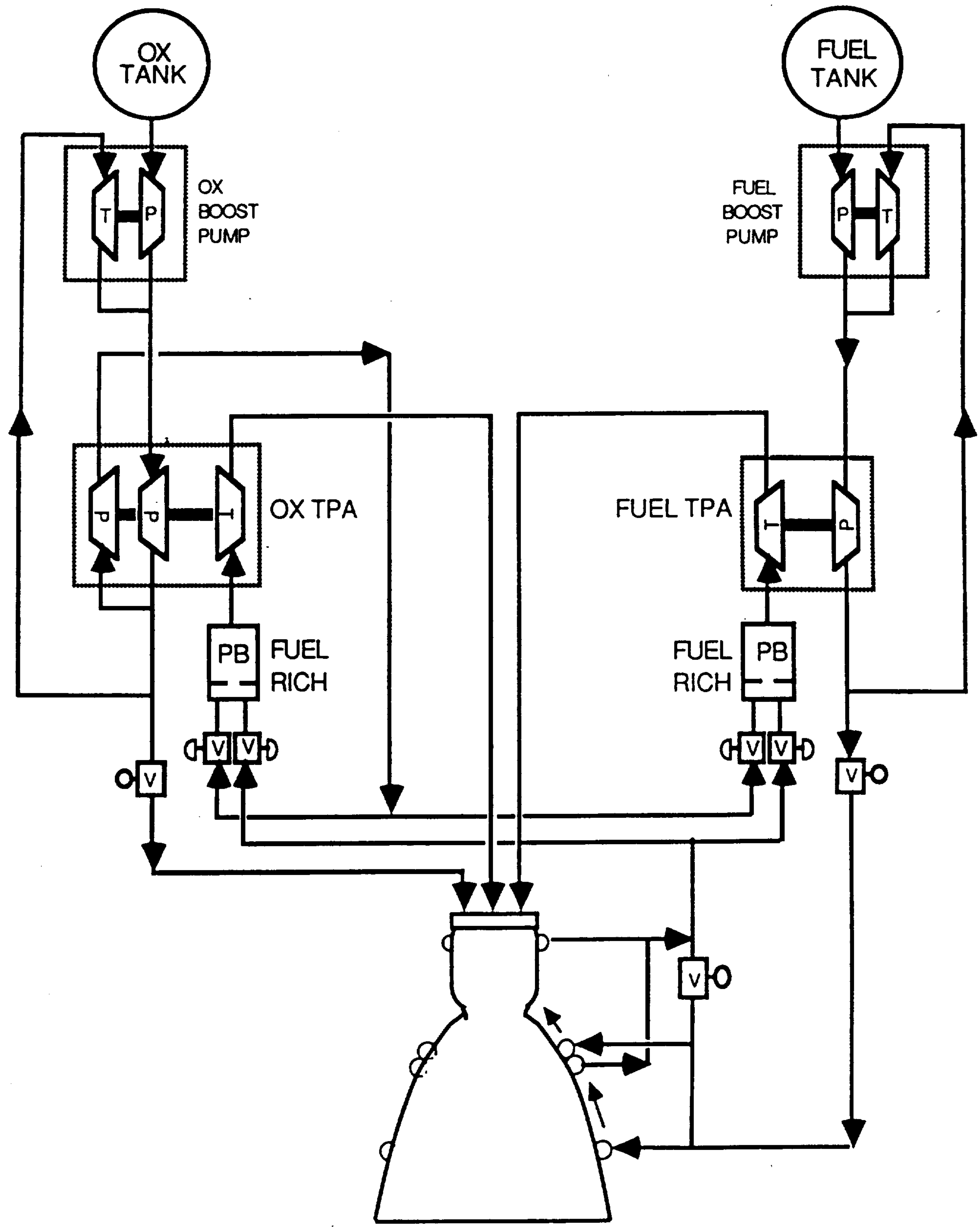

Figure 3.3-1 Power Cycle for LOX/LH2 Engine

AEROJET TECHSYSTEMS COMPANY AUG 24 1987

	CASE	P2M6E3T50	P2M7E3T50	P2M8E3T50	P2M10E3T50	P2M12E3T50	P2M14E3T50	P2M16E3T50	P2M18E3T50
	CHAMBER PRESSURE	2000.00	2000.00	2000.00	2000.00	2000.00	2000.00	2000.00	2000.00
	ENGINE THRUST (SL)	440745.	441729.	442083.	441770.	441424.	440934.	440493.	440098.
	ENGINE THRUST (VAC)	500000.	500000.	500000.	500000.	500000.	500000.	500000.	500000.
	THRUST/WEIGHT RATIO, VAC	75.44	78.07	79.16	78.43	77.17	75.84	74.52	73.25
	ENGINE WEIGHT (w/o MOUNT)	6628.	6405.	6316.	6375.	6479.	6593.	6710.	6826.
	ENGINE DEL ISP (SL)	386.85	381.81	371.61	343.93	321.03	302.06	286.01	272.18
	ENGINE DEL ISP (VAC)	438.86	432.18	420.30	389.27	363.63	342.52	324.65	309.23
	DENSITY IMPULSE (SL)	8685.	9373.	9838.	10277.	10531.	10675.	10742.	10756.
	DENSITY IMPULSE (VAC)	9853.	10610.	11127.	11632.	11929.	12104.	12193.	12220.
	ODE ISP (SL)	394.70	389.60	379.30	351.00	327.60	308.20	291.80	277.70
	ODE ISP (VAC)	446.90	440.10	428.00	396.40	370.30	348.80	330.60	314.90
	TCA ISP EFFICIENCY, SL	0.980	0.980	0.980	0.980	0.980	0.980	0.980	0.980
	TCA ISP EFFICIENCY, VAC	0.982	0.982	0.982	0.982	0.982	0.982	0.982	0.982
	NOZZLE AREA RATIO	30.00	30.00	30.00	30.00	30.00	30.00	30.00	30.00
	TCA MIXTURE RATIO	6.00	7.00	8.00	10.00	12.00	14.00	16.00	18.00
	OX FLOWRATE	976.56	1012.31	1057.46	1167.70	1269.24	1362.44	1449.53	1531.81
	FUEL FLOWRATE	162.76	144.62	132.18	116.77	105.77	97.32	90.60	85.10
	OX INLET PRESSURE	2174.	2174.	2174.	2174.	2174.	2174.	2174.	2174.
	FUEL INLET PRESSURE	2174.	2174.	2174.	2174.	2174.	2174.	2174.	2174.
	COOLANT DELTA P	1000.	1000.	1000.	1000.	1000.	1000.	1000.	1000.
	COOLANT FLOWRATE	162.76	144.62	132.18	116.77	105.77	97.32	90.60	85.10
	CSTAR	7590.71	7350.24	7102.37	6616.96	6158.05	5699.12	5242.13	4789.87
	THROAT AREA	134.40	132.15	131.31	132.08	131.59	129.29	125.47	120.36
	GG/PB CHAMBER PRESSURE	2461.	2507.	2554.	2653.	2756.	2864.	2978.	3098.
	MIXTURE RATIO	0.97	0.97	0.97	0.97	0.97	0.97	0.97	0.97
	OX FLOWRATE	43.	39.	35.	31.	28.	26.	24.	23.
Ox	FUEL FLOWRATE	45.	40.	36.	32.	29.	27.	25.	23.
PB	OX INLET PRESSURE	2895.	2949.	3005.	3121.	3242.	3369.	3504.	3644.
	FUEL INLET PRESSURE	2674.	2725.	2776.	2883.	2996.	3113.	3237.	3367.
	CSTAR	7225.	7225.	7225.	7225.	7225.	7225.	7225.	7225.
	MOLECULAR WEIGHT	3.90	3.90	3.90	3.90	3.90	3.90	3.90	3.90
	SPECIFIC HEAT RATIO	1.36	1.36	1.36	1.36	1.36	1.36	1.36	1.36
	GG/PB CHAMBER PRESSURE	2818.	2821.	2824.	2831.	2837.	2844.	2852.	2859.
	MIXTURE RATIO	0.97	0.97	0.97	0.97	0.97	0.97	0.97	0.97
	OX FLOWRATE	114.	102.	93.	82.	74.	68.	64.	60.
Fuel	FUEL FLOWRATE	118.	105.	96.	85.	77.	71.	66.	62.
PB	OX INLET PRESSURE	3315.	3319.	3323.	3330.	3338.	3346.	3355.	3364.
	FUEL INLET PRESSURE	3063.	3067.	3070.	3077.	3084.	3092.	3100.	3108.
	CSTAR	7225.	7225.	7225.	7225.	7225.	7225.	7225.	7225.
	MOLECULAR WEIGHT	3.90	3.90	3.90	3.90	3.90	3.90	3.90	3.90
	SPECIFIC HEAT RATIO	1.36	1.36	1.36	1.36	1.36	1.36	1.36	1.36
	OX PUMP INLET PRESSURE	30.	30.	30.	30.	30.	30.	30	30
	INLET TEMPERATURE	163.	163.	163.	163.	163.	163.	163.	163
	FLOWRATE	976.56	1012.31	1057.46	1167.70	1269.24	1362.44	1449.53	1531.81
	DISCHARGE PRESSURE	3543.	3547.	3551.	3559.	3567.	3576.	3585	3595
	FUEL PUMP INLET PRESSURE	4.	4.	4.	4.	4.	4.	4.	4.
	INLET TEMPERATURE	37	37.	37.	37.	37.	37.	37	37
	FLOWRATE	162.76	144.62	132.18	116.77	105.77	97.32	90.60	85.10
	DISCHARGE PRESSURE	4488.	4492.	4496.	4503.	4512	4520.	4529.	4539
	TURBINE INLET PRESSURE	2461.	2507.	2554.	2653.	2756.	2864.	2978.	3098.
	INLET TEMPERATURE	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00
Ox	FLOWRATE	88.18	78.35	71.61	63.26	57.30	52.72	49.08	46.10
PB	OUTLET PRESSURE	2185.	2185.	2185.	2185.	2185.	2185.	2185.	2185.
	HORSEPOWER	11178	11464.	11876.	12966.	13986.	14925.	15824.	16662
	TURBINE INLET PRESSURE	2818.	2821.	2824	2831.	2837.	2844.	2852	2859
	INLET TEMPERATURE	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00	1880.00
Fuel	FLOWRATE	232.46	206.55	188.79	166.78	151.07	138.99	129.39	121.54
PB	OUTLET PRESSURE	2185.	2185	2185	2185.	2185	2185.	2185	2185.
	HORSEPOWER	62019	55331	50781	45240.	41338	38381.	36070	34202

Figure 3.3-2 Typical Parametric Engine Data for LOX/LH2 Engines

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three basic burn modes: series, parallel and parallel with crossfeed. The Expendable Liquid Engine Simulation (ELES) program was utilized to evaluate the dry weight impacts on a typical reusable launch vehicle due to different pressurization system concepts. This investigation formed the basis of selecting an autogenous pressurization system for the baseline configuration.

3.3.3.2 Sizing Assumptions

The performance requirements for the selected baseline configurations dictated payload and orbit requirements. In addition, the vehicle configuration also determined many of the vehicle design parameters used in the two sizing models. However, a number of other sizing assumptions had to be made to facilitate the analysis. Some sizing parameters formed the basis for vehicle optimization, such as boost phase duration, when they would affect the possible results from the fuel trade studies. Other parameters were selected on the basis of previous sizing work.

3.3.3.3 Engine Data

High chamber pressure, 20.7 MPa, engine data was selected from the supplied LOX/LH2 engine data for a range of mixture ratios. Also selected were expansion ratios for boost phase engines that generated a 41.4 KPa exit pressure, which is consistent with past booster engine studies[1-3] and the ongoing Space Transportation Booster Engine (STBE) studies[10]. A LOX/LH2 engine with a mixture ratio of 6, consistent with current LOX/LH2 engines and projected versions in the Space Transportation Main Engine (STME) studies[11] was selected for the upper stage engine in the two stage vehicle. This engine had an expansion ratio that generated an exit pressure of 20.7 KPa, again consistent with the existing SSME. The same engine, as for the second stage of the two stage system, was selected for the sustainer phase for the SSTO except an expansion ratio of 150 was assumed for altitude operation. The sustainer phase occurs after the vehicle leaves the atmosphere so the high expansion ratio is justified. If this engine operated in parallel with other engines at lift-off then an initial expansion ratio was assumed that generated the required exit pressure, see above, and a translating nozzle was assumed to extend the expansion ratio to 150 during the sustainer phase of flight. This engine was to be used during Subtask 1.2 trade studies.

The parametric data supplied in the three contract reports for the Hydrocarbon Rocket Engine Study was examined. Conflicting trends between the three reports were found. For example, Pratt and Whitney showed parametric data that indicated that engine thrust to weight went up as engine thrust level went up, an exact opposite to the trends reported by the other participants. The AeroJet TechSystems report was finally used as the source of engine data as it was the most consistent and had the range of data needed. From reviewing all the reports, chamber pressures for the fuel cooled engines that limited fuel pump discharge pressures to below 51.8 MPa were selected. A constant chamber pressure of 20.7 MPa for the hydrogen cooled engines was used. Engine performance and near term specific impulse performance associated with the selected chamber pressures values were then identified from the AeroJet parametric data. This was done for a range of thrust levels. The engines selected all had an expansion ratio that generated an exit pressure of 41.4 KPa. Note that all of the engines selected utilized a gas generator power cycle.

3.3.4 Baseline Vehicle Results and Conclusions

3.3.4.1 SSTO

General

The geometry for this baseline configuration is shown in Figure 3.3-3. This vehicle has a payload capacity of 13,605 kgs delivered to the low earth orbit of 93 by 186 kilometers at 28 degrees inclination from an ETR launch; final orbit is achieved using the orbital maneuvering system (OMS). The maximum acceleration is limited to 3.5 g's due to the presence of the flight crew. The payload bay size is 4.25 meters in diameter by 19 meters in length. These performance parameters were determined by the original configuration design[8]. The vehicle subsystems are described below.

Subsystem Descriptions

Propulsion

The main propulsion system for the boost phase is either an LOX/LH2 engine or a LOX/hydrocarbon engine. Pressurization is assumed to be autogenous. Feed, fill and drain lines will use insulated rather than vacuum-jacketed lines for cryogenic propellants.

The baseline vehicle from Reference 8 uses an integrated system for the secondary propulsion subsystems and power. This requires that the OMS, the auxiliary power system (APS) and the reaction control system (RCS) all use the same fuel and oxidizer supplied by the OMS tanks. Because of the design complexity of retaining this concept of integrated subsystems when fuels change, it is assumed that, when the main engine fuel changes from hydrogen, a separate hydrogen tank, gaseous hydrogen and oxygen thrusters, LOX/LH2 OMS engines and a standard GOX/GH2 power system must be retained for the integrated, secondary propulsion and power systems. This will result in a weight and cost penalty for non-hydrogen fueled vehicles, but it should be of minor impact.

The main propulsion engines are pump-fed and have an assumed life of 100 missions. Gimballing is used for thrust vector control and is provided by hydraulics.

Other

The vehicle selected employs vertical takeoff with a dead stick horizontal landing. The vehicle uses a automated configuration control system to maintain aerodynamic stability.

The airframe structure is an un-pressurized structure combined with integrated tankage. The tankage system for this vehicle employs a welded titanium honeycomb sandwich with ring stiffened sidewalls for the fuel tank and an aluminum 2219 alloy in a skin stringer construction for the oxidizer tank. As the original vehicle was assumed to use a dual fuel engine, the secondary fuel (methane in the case of the original system) tank is placed above the oxidizer tank and shares a common dome with the latter. This tankage arrangement will be altered, when the engine fuels change, in appropriate manner. Internal insulation is used as required. Separate accumulator tanks are allocated for the secondary propulsion subsystems.

The primary and secondary structures employ organic and metal composites. The vehicle secondary structures include the crew module, payload bay, aft body area and the body flap. Reusable surface insulation over composite standoff panels are used for thermal protection on all surfaces.

The provided aerosurfaces include the main wing, deployable canards, the deployable yaw ventral, the wing tiplets and the body flap. All of these are constructed from organic composites. Aerosurfaces were designed for minimum area consistent with landing requirements.

The power system uses APUs, auxiliary batteries and alternators.

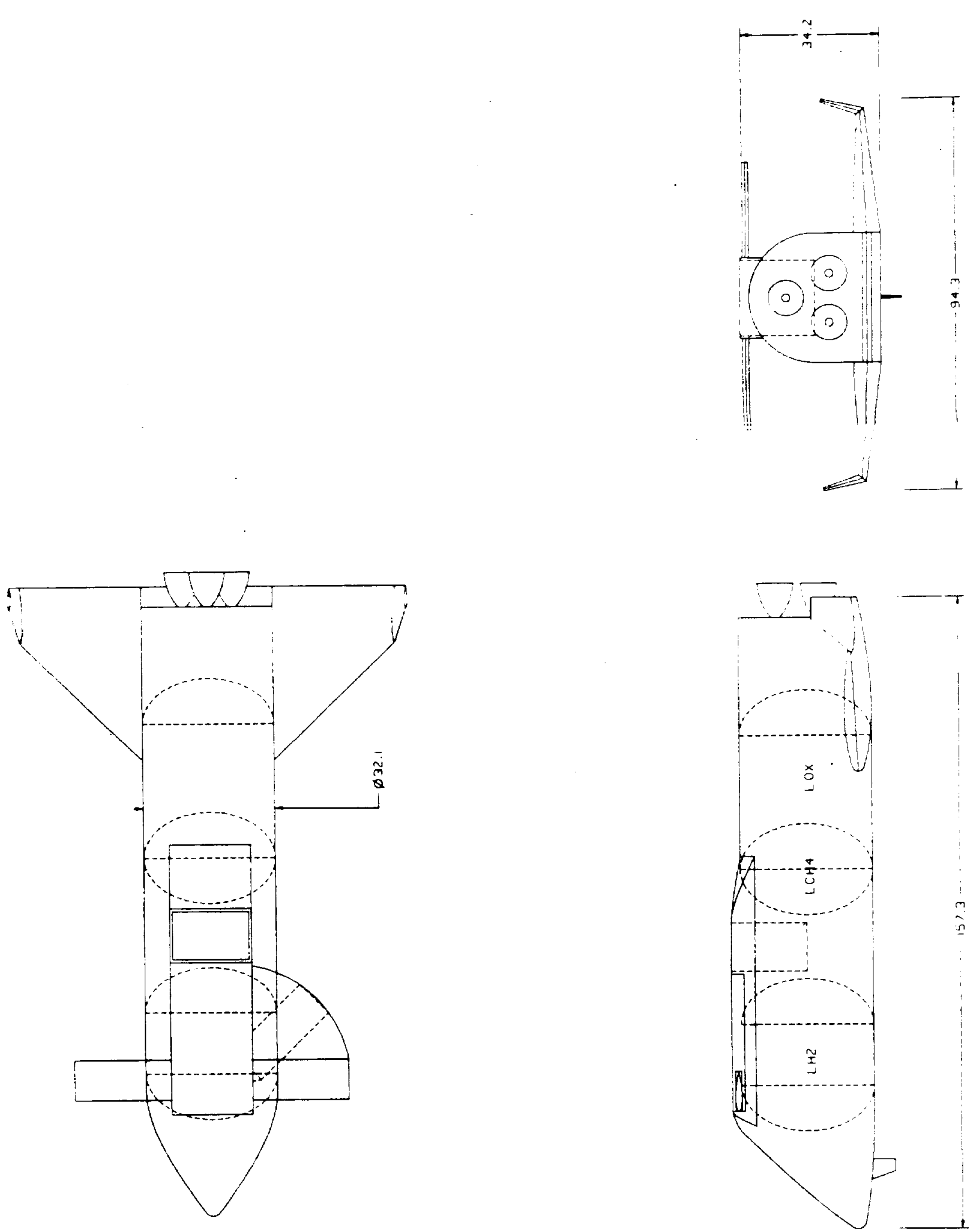


Figure 3.3-3 SSTO Baseline Configuration

3.3.4.2 Two-Stage

General

The geometry for this system is shown in Figure 3.3-4. This vehicle is designated the unmanned fully reusable cargo vehicle (UFRCV). This vehicle has a payload capacity of 29,478 kgs delivered to the low earth orbit of 105 by 280 Km at 28 degrees inclination from an ETR launch. The maximum acceleration is limited to 4.5 g's which is consistent with NASA guidelines for advanced, unmanned cargo vehicle designs. The payload bay size is 4.57 meters in diameter by 19.8 meters in length. The payload bay size and orbit were determined after an analysis of the STAS mission model data base.

The vehicle uses a parallel burn of both stages at lift-off. This burn mode was selected on the basis of the burn mode investigation. The burn mode analysis was conducted for series burn, parallel burn and parallel burn with crossfeeding vehicles using different fuel combinations in the booster of the UFRCV. A range of payload delivery orbits were investigated. It was found that the three burn types exhibited similar sensitivity to destination orbit. Efforts were focused on the recommended low earth orbit of 105 by 280 Km. For this orbit, the parallel and cross-fed configurations were generally lower in total dry weight than the series burn case, but not more than 10 percent as is shown in Figures 3.3-5 and 6 for a LOX/CH_4 and LOX/LH_2 booster respectively. The series and cross-fed configurations were slightly more sensitive to fuel type than the parallel burn mode. The former two cases exhibited a 15 percent change in total vehicle dry mass, versus a reference, when the fuel was changed from hydrogen to methane, see Figure 3.3-7 which shows the optimum points on the respective curves of Figures 3.3-5 and 6. The parallel burn cases only showed a 12 percent sensitivity to fuel type. The cross-fed configuration for the methane case had the lowest total dry weight, but only 2 percent lower than the parallel burn mode. The parallel burn mode for the hydrogen case was slightly, on the order of one-half percent of the reference dry mass, better than the cross fed case.

On the basis of the above, and previous work, a parallel burn mode for the baseline vehicle was selected. This burn mode has a greater weight efficiency than the series burn mode and almost as good as that for the cross-fed burn mode, which will be studied later in any case. Its sensitivity to the propulsion options to be studied later is only slightly less than that for the series mode. Finally, this burn mode lends itself to both Subtask 1.3, which examines cross-feeding propellants while burning in parallel, and Subtask 1.5, which will examine two position nozzles, a concept that makes most sense for a parallel burn vehicle.

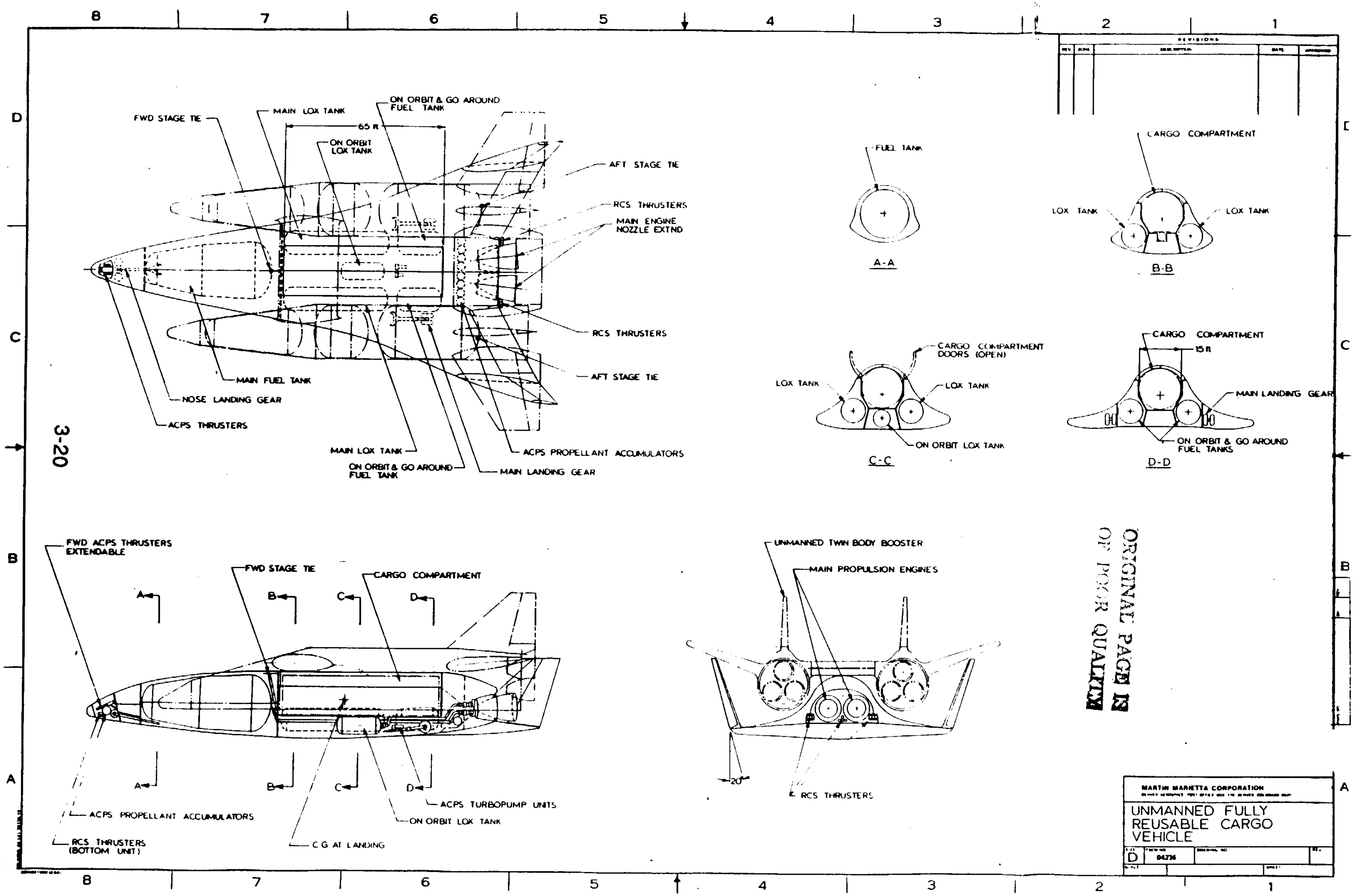


Figure 3.3-4 UFRCV Baseline Configuration

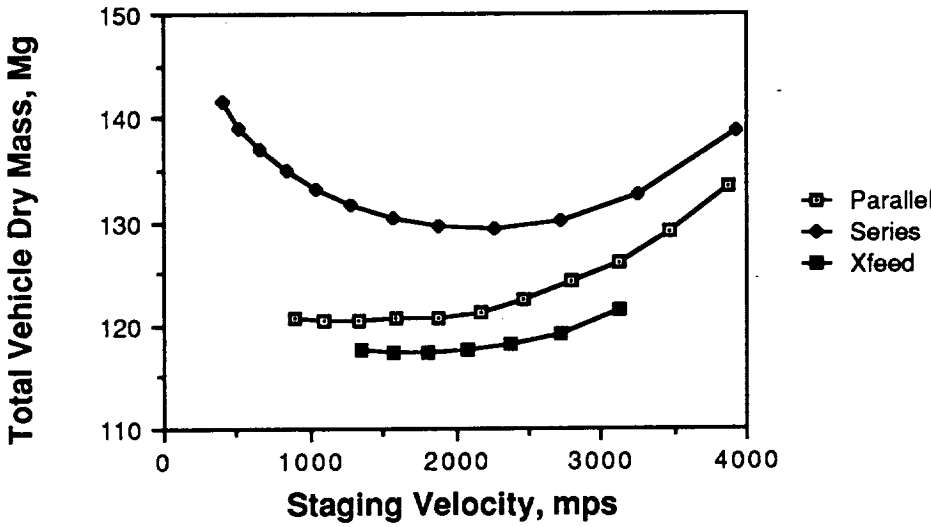


Figure 3.3-5 Total UFRCV Dry Mass Versus Burn Mode - LOX/Methane Booster

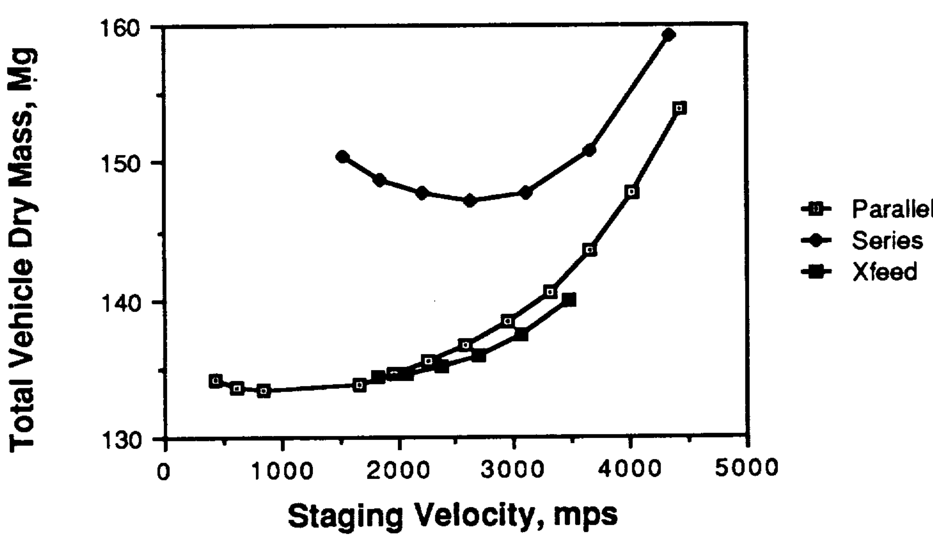


Figure 3.3-6 Total UFRCV Dry Mass Versus Burn Mode - LOX/Hydrogen Booster

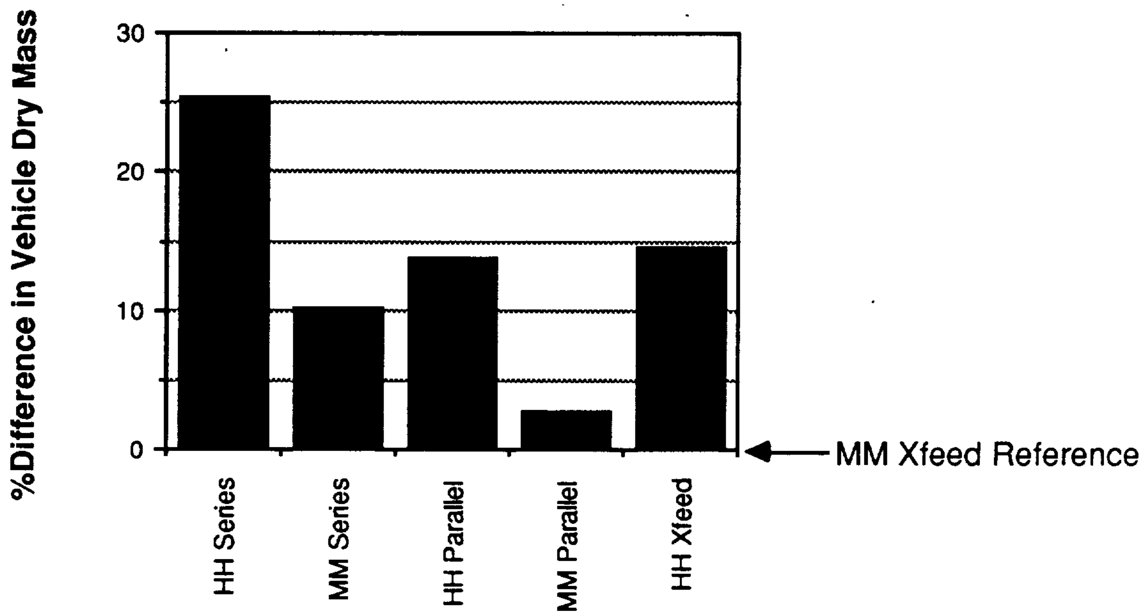


Figure 3.3-7 Comparison of Different Burn Modes and Fuels for UFRCV, Reference is Methane Booster Using Crossfeed to Methane Orbiter

Subsystem Descriptions

Propulsion Subsystems:

Certain propulsion subsystems were set by the nature of the study. These include: pump-fed engines, integral tankage, and internal tankage insulation. Other propulsion subsystems were determined through a combination of sizing, using WASP, and option evaluation using ELES. It was found that different options for feed systems, materials and type of construction, had little impact on the vehicle size; so insulated feedlines, using state of the art materials, were selected. Vehicle, and subsystem, weights varied significantly for the two different pressurization schemes, autogenous and helium blow down. Although the autogenous pressurization required greater amounts of gas, fuel and oxidizer, the helium storage tank was such a significant weight item that autogenous pressurization is preferred. Table 3.3-1 shows a comparison between the two schemes for a LOX/CH4 booster, all weights are in kgs. However, when RP-1 was used as an engine fuel and coolant it was assumed that a blow down pressurization scheme would be used due to the inability to properly generate autogenous gas from RP-1. This generated an added weight penalty in the trades analysis when the RP-1 fuel/coolant option was examined.

Table 3.3-1 Autogenous vs Helium Pressurization

Pressurization Method Comparison (units in kgs)		
	Autogenous	Helium
Control Hardware	1845	1873
Gaseous Oxygen	1242	n/a
Gaseous Fuel	582	n/a
Gaseous Helium	n/a	472
Heat Exchanger	140	n/a
Pressurant Tank	n/a	3288
Sum	3809	5633

Other Subsystems:

The vehicle selected employs vertical takeoff with horizontal landing for both stages; thus the aerosurfaces for both stages. The booster has flyback airbreathing engines in its forward wing while the orbiter is unpowered for return.

Figures 3.3-8 and 3.3-9 indicate the significant subsystems, other than main propulsion, on the two stages.

3.3.4.3 Sizing Ground Rules

Various sizing ground rules were determined by the selected vehicle configurations. These included performance requirements, key geometry constraints, subsystem sizing parameters etc. However, several assumptions were still necessary to conduct the sizing analyses.

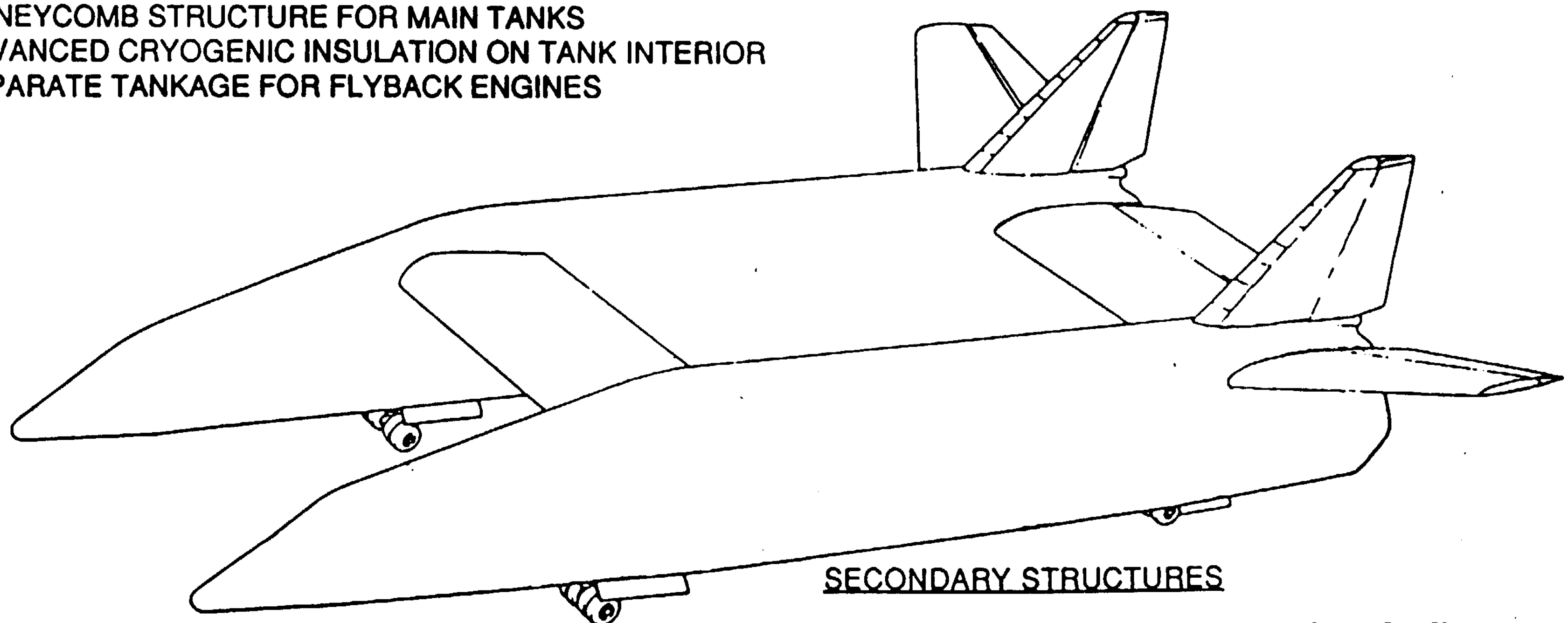
In keeping with the parametric nature of this study it was decided to allow the sizing programs to determine the number of engines required for a propulsion phase or stage on a fractional basis. Typically then, the number of engines is determined based upon dividing the single engine thrust into the total thrust required. This eliminated possible discontinuities in the sizing analysis caused by varying either the number of engines or their thrust level. In making this decision the single engine thrust levels to be used by the sizing programs during the boost phase also had to be identified.

PROPULSION SYSTEMS

- REUSABLE ENGINES WITH ADDITIONAL MARGINS
- HIGH BYPASS FLYBACK ENGINES IN FWD WING SECTION
- NEW GOX/GFUEL RCS THRUSTERS ALSO PROVIDE SEPARATIOI

STRUCTURES AND TANKAGE

- AL-LI ALLOY SKIN PANELS ON TITANINUM CORE HONEYCOMB STRUCTURE FOR MAIN TANKS
- ADVANCED CRYOGENIC INSULATION ON TANK INTERIOR
- SEPARATE TANKAGE FOR FLYBACK ENGINES



TPS

- HONEYCOMB RADIATIVE SURFACE PANELS USE DIFFERENT MATERIALS BASED UPON TEMP.
- ADV. HIGH TEMP INSULATION UNDER SURFACE PANELS TO PROTECT TANKAGE
- HOT STRUCTURE USED IN CONTROL SURFACES

SECONDARY STRUCTURES

- ORGANIC COMPOSITES USED FOR NON-TANK SECTIONS
- HIGH TEMPERATURE (~500°) ADVANCED ALUMINUM ALLOY USED FOR RIBS AND STRINGERS

OTHER SYSTEMS

- STAGE AVIONICS PROVIDE COMPLETE AUTONOMOUS OPERATION
- POWER PROVIDED BY LOX/FUEL POWER TURBINES
- UNITARY HYDRAULICS USED FOR MAIN ENGINE GIMBALING

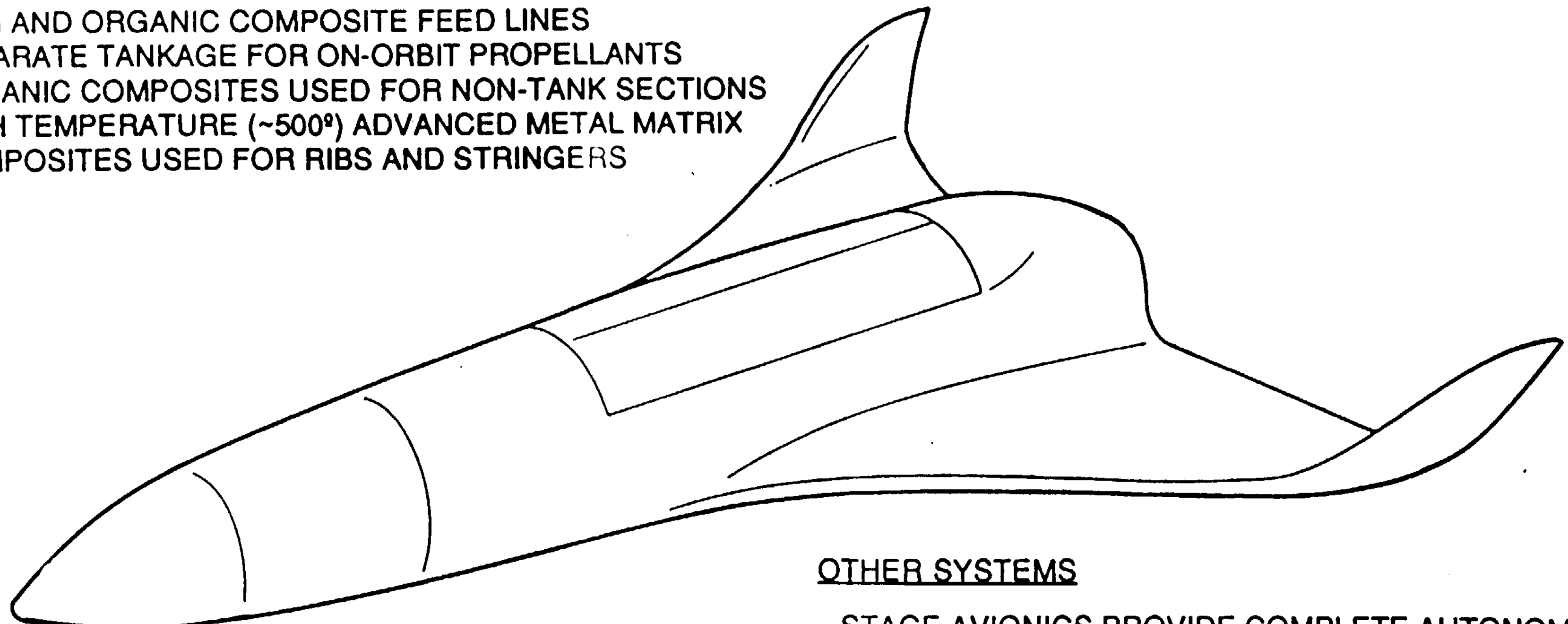
Figure 3.3-8 Descriptions of Subsystems for UFRCV Booster

STRUCTURES AND TANKAGE

- NON-INTEGRAL TANKAGE USES AL-LI ALLOY SKIN PANELS ON TITANINUM CORE HONEYCOMB STRUCTURE FOR MAIN TANKS
- AL-LI AND ORGANIC COMPOSITE FEED LINES
- SEPARATE TANKAGE FOR ON-ORBIT PROPELLANTS
- ORGANIC COMPOSITES USED FOR NON-TANK SECTIONS
- HIGH TEMPERATURE (~500º) ADVANCED METAL MATRIX COMPOSITES USED FOR RIBS AND STRINGERS

PROPULSION SYSTEMS

- NEW ADVANCED LOX/LH2 ENGINE WITH 2 PSN NOZZLE AND ON-ORBIT RESTART
- ENGINE HAS ON-BOARD HEALTH MONITORING
- ALL AUXILARY PROPULSION SYSTEMS USE MAIN ENGINE PROPELLANTS



TPS

- HONEYCOMB RADIATIVE SURFACE PANELS USE DIFFERENT MATERIALS BASED UPON TEMP.
- ADV. HIGH TEMP INSULATION UNDER SURFACE PANELS TO PROTECT TANKAGE
- HOT STRUCTURE USED IN MAIN WING AND CONTROL SURFACES
- ACTIVE TPS LIMITS MATERIAL MAXIMUM TEMPERATURE IN HOT STRUCTURE AREAS

OTHER SYSTEMS

- STAGE AVIONICS PROVIDE COMPLETE AUTONOMOUS OPERATION
- POWER PROVIDED BY LOX/LFUEL POWER TURBINES
- UNITARY HYDRAULICS USED FOR MAIN ENGINE GIMBALING

Figure 3.3-9 Descriptions of Subsystems for UFRCV Orbiter

For the SSTO, the selection of thrust level was based on typical LOX/LH2 engines and a preliminary sizing analysis. Through preliminary sizing for the SSTO it was found that selecting a high thrust versus a low thrust LOX/LH2 engine had little significant impact on the total vehicle dry weight. The use of higher thrust engines generated vehicles less that 1% heavier, in total dry weight, than the use of lower thrust engines. Since the use of higher thrust engines was not justified it was decided that a thrust level of 2.2 MN (vacuum) for the LOX/LH2 engine would be used. This value was the lowest thrust level available from the supplied data and is consistent with the thrust level of the SSME and close to that recommended for the STME. For the LOX/hydrocarbon engines on the SSTO a thrust level of approximately 3.1 MN (vacuum) was selected. This value resulted in a reasonable number of engines, somewhere between 3-6, for the SSTO parallel burn sizing and is close to that used for the STBE configuration studies.

For the UFRCV the booster engine thrust level for both LOX/LH2 engines and LOX/hydrocarbon engines was set at approximately 3.1 MN (vacuum). This thrust level, as noted earlier, is consistent with that considered in the STBE studies. The LOX/LH2 boost engines were forced to be at the same thrust level as the LOX/Hydrocarbon in order to provide a better basis of comparison between the reference vehicle and subsequent configurations using LOX/Hydrocarbon engines.

Other performance ground rules had to be established for vehicle sizing besides engine thrust level. A vehicle thrust to weight ratio at lift-off of 1.3 was assumed for both vehicles. This was selected as being large enough to limit gravity losses but not excessive enough to require substantial engine throttling during flight to avoid exceeding maximum acceleration limits. It was also necessary to establish a vehicle thrust to weight ratio at staging for the UFRCV and at end of boost phase for the SSTO. How this was done varied between the vehicle types. For the SSTO, the assumed vehicle thrust to weight at end of boost phase was assumed to be 1.0 for the series burn concept. For the parallel burn SSTO a constant hydrogen mass flow rate for the sustainer engine was assumed. Thus, the vehicle thrust to weight ratio at the end of the boost phase would be dictated by the initial value for the thrust fraction, described below. As in the case of the parallel burn SSTO , the parallel burn UFRCV has its vehicle thrust to weight ratio at staging established by the thrust ratio value selected, also described below. In these latter two cases, the optimization of the vehicle on the basis of thrust fraction, or ratio, also optimized for vehicle thrust to weight ratio.

Optimization parameters for vehicle sizing were selected for both configurations. These parameters are to be varied across a range in order to select an optimum vehicle design. For the SSTO, a burn type, which is either parallel or series, is also selected. The former is defined as when the hydrocarbon engines are burned in parallel with the hydrogen (sustainer) engines at lift-off. The series burn mode is defined as when the hydrocarbon engines burn first during the boost phase and then stop when the hydrogen engines burn during sustainer phase. For the SSTO parallel burn mode an additional optimization parameter is the fraction of total lift-off thrust that is provided by the boost phase engines, this is called the thrust fraction. Both burn modes of the SSTO, and the parallel burn UFRCV also use boost phase duration, which dictates staging velocity for the UFRCV, as an optimization parameter. For the parallel burn UFRCV an additional optimization parameter is the thrust ratio, defined as the ratio of total stage two thrust to total stage one thrust.

3.3.4.4 Engine Data

The engine data selected for LOX/LH2 engines over a mixture ratio range is shown in Tables 3.3-2 through 4. Table 3.3-2 is for boost phase engines, all of which have an expansion ratio that generates an exit pressure of 41.4 KPa. Table 3.3-3 is the data used for the SSTO sustainer phase engine. All are assumed to have an expansion ratio that generates an exit pressure of 41.4 KPa at lift-off and then extend a translating nozzle to obtain an expansion ratio of 150 at boost phase termination. Thus the weights include the translating nozzle while the specific impulses are reported as sea level and vacuum performance while in boost mode and vacuum performance for sustainer mode. Note the range of thrusts. Table 3.3-4 is the data used for the second stage engine for the UFRCV. Each engine is assumed to have a constant expansion ratio that generates an exit pressure of 20.7 KPa.

The hydrocarbon engines to be used during SSTO boost phase or on the booster of the UFRCV are shown in Tables 3.3-5 and 3.3-6. Table 3.3-5 is for fuel cooled engines while Table 3.3-6 is for hydrogen cooled engines. Note that the chamber pressure for the hydrogen cooled engines is assumed to be 20.7 MPa for all fuels while the chamber pressure varies for the fuel cooled engines. The values of %LH2 represent the percentages of total propellant flow to the engine that is hydrogen. All of the listed values for specific impulse are for near term performance, the performance that is believed obtainable in the next three to five years.

3.4 Task 1.2 - Establish Reference Vehicles, Conduct Propellant Trades and Sensitivities

3.4.1 Objective

The objectives of this subtask were to: (a) establish a reference vehicle design for each of the two vehicle baseline configurations using LOX/LH2 engines and (b) conduct a trade study analysis to determine vehicle designs for the SSTO and UFRCV using the different hydrocarbon engine options, listed in Table 3.4-1.

Table 3.3-2 Boost Phase LOX/LH2 Engines

Staged Combustion
Chamber Pressure = 20.7 MPa (3000 psi)
Exit Pressure = 41.4 KPa (6.0 psi)
Expansion Ratio = 41.6:1

MR = 6

Fvac	Isp vac	T/Weng
2.2 MN	445.62	79.06
3.3 MN	"	78.08
4.4 MN	"	76.92

MR = 7

Fvac	Isp vac	T/Weng
2.2 MN	439.92	82.25
3.3 MN	"	81.16
4.4 MN	"	79.93

MR = 8

Fvac	Isp vac	T/Weng
2.2 MN	429.07	83.84
3.3 MN	"	82.65
4.4 MN	"	81.13

MR = 10

Fvac	Isp vac	T/Weng
2.2 MN	396.57	83.21
3.3 MN	"	81.68
4.4 MN	"	80.15

MR = 12

Fvac	Isp vac	T/Weng
2.2 MN	369.76	81.95
3.3 MN	"	80.29
4.4 MN	"	78.61

MR = 14

Fvac	Isp vac	T/Weng
2.2 MN	347.85	80.52
3.3 MN	"	78.66
4.4 MN	"	76.86

MR = 16

Fvac	Isp vac	T/Weng
2.2 MN	329.43	79.04
3.3 MN	"	77.01
4.4 MN	"	75.10

MR = 18

Fvac	Isp vac	T/Weng
2.2 MN	313.65	77.83
3.3 MN	"	75.41
4.4 MN	"	73.40

Table 3.3-3 Sustainer Phase LOX/LH2 Engines

Staged Combustion
Chamber Pressure = 20.7 MPa (3000 psi)
Expansion Ratio = 150:1

MR = 6

Fvac	Isp vac	T/Weng
2.2 MN	466.35	64.66
3.3 MN	"	64.07
4.4 MN	"	63.36

MR = 7

Fvac	Isp vac	T/Weng
2.2 MN	463.11	67.37
3.3 MN	"	66.75
4.4 MN	"	65.98

MR = 8

Fvac	Isp vac	T/Weng
2.2 MN	454.76	68.94
3.3 MN	"	68.27
4.4 MN	"	67.45

MR = 10

Fvac	Isp vac	T/Weng
2.2 MN	418.14	68.13
3.3 MN	"	67.29
4.4 MN	"	66.36

MR = 12

Fvac	Isp vac	T/Weng
2.2 MN	388.58	67.14
3.3 MN	"	66.15
4.4 MN	"	65.11

MR = 14

Fvac	Isp vac	T/Weng
2.2 MN	364.42	66.30
3.3 MN	"	65.17
4.4 MN	"	64.02

MR = 16

Fvac	Isp vac	T/Weng
2.2 MN	344.19	65.60
3.3 MN	"	64.33
4.4 MN	"	63.09

MR = 18

Fvac	Isp vac	T/Weng
2.2 MN	326.91	65.08
3.3 MN	"	63.66
4.4 MN	"	62.32

Table 3.3-4 Second Stage LOX/LH2 Engines

Staged Combustion
Chamber Pressure = 20.7 MPa (3000 psi)
Exit Pressure = 20.7 KPa (3.0 psi)
Expansion Ratio = 70:1

MR = 6

Fvac	Isp vac	T/Weng
2.2 MN	455.16	75.24
3.3 MN	"	
4.4 MN	"	

MR = 7

Fvac	Isp vac	T/Weng
2.2 MN	450.54	78.41
3.3 MN	"	77.49
4.4 MN	"	76.58

MR = 8

Fvac	Isp vac	T/Weng
2.2 MN	440.62	80.10
3.3 MN	"	79.10
4.4 MN	"	77.93

MR = 10

Fvac	Isp vac	T/Weng
2.2 MN	406.35	79.35
3.3 MN	"	78.13
4.4 MN	"	76.80

MR = 12

Fvac	Isp vac	T/Weng
2.2 MN	378.46	78.13
3.3 MN	"	76.71
4.4 MN	"	75.25

MR = 14

Fvac	Isp vac	T/Weng
2.2 MN	355.58	76.86
3.3 MN	"	75.26
4.4 MN	"	73.68

MR = 16

Fvac	Isp vac	T/Weng
2.2 MN	336.33	75.62
3.3 MN	"	73.85
4.4 MN	"	72.16

MR = 18

Fvac	Isp vac	T/Weng
2.2 MN	319.84	74.46
3.3 MN	"	72.53
4.4 MN	"	70.73

Table 3.3-5 LOX/HC/Fuel Cooled Engine Data

Gas Generator Cycle
Exit Pressure = 41.4 KPa (6.0 psi)
Chamber Pressure = (dependant on fuel)
Hydrocarbon Cooled
Near Term Performance (3-5 years)

LO2/RP-1

PC = 10.3 MPa (1500 psi)
MR =2.42 AR = 28.7

F (S.L.)	Isp (S.L./Vac)	T/Weng
1.1 MN	264.5/312.0	117.0
3.2 MN	"	103.3
5.3 MN	"	94.7
10.6 MN	"	79.1

LO2/CH4

PC = 20.7 MPa (3000 psi)
MR =3.05 AR = 48.2

F (S.L.)	Isp (S.L./Vac)	T/Weng
1.0 MN	306.3/350.0	121.3
3.1 MN	"	111.3
5.2 MN	"	103.7
10.4 MN	"	89.0

LO2/C3H8(NBP)

PC = 18.6 MPa (2700 psi)
MR =2.62 AR = 45.5

F (S.L.)	Isp (S.L./Vac)	T/Weng
1.1 MN	287.0/330.2	115.5
3.1 MN	"	109.0
5.2 MN	"	102.2
10.5 MN	"	87.8

LO2/C3H8(SC)

PC = 20.7 MPa (3000 psi)
MR =2.70 AR = 49.0

F (S.L.)	Isp (S.L./Vac)	T/Weng
1.0 MN	290.7/332.8	131.9
3.1 MN	"	121.3
5.2 MN	"	113.0
10.4 MN	"	96.8

Table 3.3-6 LOX/HC/H2 Cooled Engine Data

Gas Generator Cycle
Exit Pressure = 41.4 KPa (6.0 psi)
Chamber Pressure = 20.7 MPa (3000 psi)
Hydrogen Cooled
Near Term Performance (3-5 years)
%LH2 is Percentage of Total Flow that is Hydrogen

LO2/RP-1/LH2

MR =2.82 AR = 48.4

F (S.L.)	Isp (S.L./Vac)	T/Weng	%LH2
1.1 MN	294.6/335.4	129.8	1.96
3.1 MN	"	122.3	1.01
5.1 MN	"	115.1	0.82
10.2 MN	"	99.5	0.67

LO2/CH4/LH2

MR =3.53 AR = 48.2

F (S.L.)	Isp (S.L./Vac)	T/Weng	%LH2
1.1 MN	316.2/359.8	121.8	2.11
3.1 MN	"	114.7	1.09
5.1 MN	"	108.1	0.88
10.2 MN	"	94.1	0.72

LO2/C3H8(NBP)/LH2

MR =3.13 AR = 49.2

F (S.L.)	Isp (S.L./Vac)	T/Weng	%LH2
1.1 MN	302.3/344.9	125.0	2.02
3.1 MN	"	117.7	1.04
5.1 MN	"	110.9	0.84
10.2 MN	"	96.0	0.69

LO2/C3H8(SC)/LH2

MR =3.13 AR = 49.0

F (S.L.)	Isp (S.L./Vac)	T/Weng	%LH2
1.1 MN	301.6/343.9	130.6	2.01
3.1 MN	"	123.2	1.04
5.1 MN	"	116.1	0.84
10.2 MN	"	100.4	0.69

Table 3.4-1 Hydrocarbon Engine Options

NBP = Normal Boiling Point; SC = Subcooled

Candidate	Fuel	Engine Coolant
1	RP-1 (R)	RP-1 (R)
2	RP-1 (R)	Hydrogen (H)
3	Methane (M)	Methane (M)
4	Methane (M)	Hydrogen (H)
5	Propane (NBP) (NP)	Propane (NBP) (NP)
6	Propane (NBP) (NP)	Hydrogen (H)
7	Propane (SC) (SP)	Propane (SC) (SP)
8	Propane (SC) (SP)	Hydrogen (H)

3.4.2 Summary of Task Activity

3.4.2.1 Reference Vehicles

The reference vehicle designs using LOX/LH2 engines were found by conducting vehicle sizing analysis using the established ground rules and baseline vehicle designs and then varying the mixture ratio for the LOX/LH2 engines. The mixture ratio range was from 6 to 8. The optimum configurations, which became the reference vehicles for the remainder of the study, were identified by selecting the systems with the lowest total vehicle dry weight.

3.4.2.2 Trade Studies

The trade study analysis was conducted by using the different engine parameters for the hydrocarbon engine options as input to the vehicle sizing analysis. Vehicle optimization was conducted on the basis of the optimization parameters discussed earlier. The optimum configurations, for both the SSTO and UFRCV, were identified for the eight hydrocarbon engine options. The optimum configurations were compared to the reference, all hydrogen, configurations for both the SSTO and the UFRCV.

3.4.2.3 Sensitivities

A sensitivity analysis to three key parameters: (a) engine thrust to weight, (b) engine mixture ratio and (c) engine specific impulse was also conducted. The sensitivity analysis for specific impulse spans a range that includes the far term performance, believed obtainable in ten years, of the hydrocarbon engine options as defined in Reference 1.

3.4.3 Discussion of Analysis Procedure

3.4.3.1 Ground Rules and Assumptions Used

Reference Vehicles

The ground rules for sizing and sizing optimization parameters used for defining the reference vehicles were established in Subtask 1.1. For the SSTO reference vehicle analysis, the assumption was made that the boost phase and sustainer phase of flight was generated by one engine that operated at a low expansion ratio at lift-off and shifted expansion ratio after the boost phase was over. This implies that the selected mixture ratio is used for the entire vehicle flight. This contrasts with the assumption used for the UFRCV. For this vehicle, It was assumed that the booster engine would be varied, based upon mixture ratio, while a single version of a LOX/LH2 engine was used in the second stage throughout the analysis.

Although engine data was obtained for a range of thrust levels in Subtask 1.1, a constant thrust level for the LOX/LH2 engines was used in the reference vehicle analysis for both the SSTO and UFRCV, as previously described in Section 3.3.4.3.

Trade Studies

As for the reference vehicle analysis, the general sizing ground rules were used for the trade studies and sensitivity analyses. As noted in Section 3.3.4.3, it was assumed that two burn types were possible for the SSTO trades, series and parallel. It was further assumed that the sustainer phase engine for the SSTO was a LOX/LH2 engine operating at mixture ratio 6 and a vacuum thrust level of 2.2 MN. This same engine was used in the UFRCV upper stage, although with a fixed nozzle, while the booster engines were varied during the trade study.

Sensitivities

The ranges of the sensitivity analyses for the three parameters were limited to values that would be large enough to show sensitivities but small enough to be considered reasonable. In addition, for specific impulse, the range to values were limited to those necessary to capture the far term specific impulse performance values for the hydrocarbon fuels. Unfortunately, the vehicle sensitivities to mixture ratio were very slight even over the extreme range selected. The ranges selected for the three parameters: specific impulse, mixture ratio and engine thrust to weight, are shown in Table 3.4-2. Note that methane has a smaller range than the other hydrocarbon fuels because the far term specific impulse performance for this fuel is not much different from near term performance.

Table 3.4-2 Sensitivity Ranges

Specific Impulse Range	-3% to +3% for all fuels except Methane -1% to +1% for methane fueled engines
Mixture Ratio	-50% to + 50% for all fuels
Engine Thrust to Weight	-15% to + 15% for all fuels

The sensitivity analyses were conducted for only the optimum vehicles found during the trade studies for hydrocarbon fueled engines in the SSTO and UFRCV. This implies that no variation of the parameters used for optimizing the vehicles is necessary.

3.4.3.2 Input Data

Reference Vehicles

SSTO

Other than the baseline vehicle characteristics and sizing ground rules, the only input data required was for the LOX/LH2 engines at the different mixture ratios. This was readily available from the supplied LOX/LH2 parametric data. The specific engine data used for this analysis is shown in Table 3.4-3.

UFRCV

The input data for the UFRCV included the standard sizing ground rules, the baseline vehicle characteristics and LOX/LH2 booster engine data for the different mixture ratios. This engine data is shown in Table 3.4-4.

Table 3.4-3 LOX/LH2 Engine Data for SSTO Reference Analysis

MR	Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
6.00	445.62/466.35	41.6/150	20.7	2224	3508	2.41/8.20
7.00	439.92/463.11	41.6/150	20.7	2224	3366	2.36/8.00
8.00	429.07/454.76	41.6/150	20.7	2224	3290	2.35/7.89

Table 3.4-4 LOX/LH2 Booster Engine Data for UFRCV Reference Analysis

MR	Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
6.00	445.62	41.6	20.7	3336	2869	2.41
7.00	439.92	41.6	20.7	3336	2757	2.36
8.00	429.07	41.6	20.7	3336	2705	2.35

Trade Studies

SSTO

As noted earlier, the point thrust level used for the LOX/Hydrocarbon engines was to be approximately 3100 KN. This thrust level resulted in a reasonable number of hydrocarbon engines on the optimized, parallel burn vehicles (3 or 4 engines). However, the point thrust level for the the series burn analysis was adjusted to be approximately 5200 KN. If thrust values of 3100 KN had been used the optimized, series burn vehicles would have required from 8 to 13 engines. Therefore, the larger thrust value of 5200 KN was selected because it resulted in a more reasonable number of hydrocarbon engines, from 5 to 8 on the optimized vehicles. It should be noted that the use of even higher thrust values, such as 10400 KN, would have resulted in even fewer engines but a heavier vehicle because of the lower thrust to weight inherent in the higher thrust engines.

It will be seen in the results that the series burn vehicles have considerably greater dry mass than the parallel burn vehicles and therefore the selection of hydrocarbon engine thrust level is not a crucial issue for this analysis.

Tables 3.4-5 and 3.4-6 show the exact engine data used for the parallel and series burn analysis respectively for the eight hydrocarbon engine options. Note that the engine thrust levels in each table are not the same. This is a result of the source of the engine parametric data, Reference 1. The available data was constrained to certain specific thrust levels for the different hydrocarbon engine options and the thrust levels were not always the same for each hydrocarbon.

UFRCV

The engine data used for the UFRCV trade studies was the same as that for the parallel burn SSTO trades, although a constant thrust level of 3,336.3 KN was used by curve fitting the parametric data. Table 3.4-7 shows the exact engine data used.

Sensitivities

The input for the sensitivity analyses for both the SSTO and UFRCV was the optimum vehicle configurations for the eight hydrocarbon engine options and the parameter ranges to be investigated.

Table 3.4-5 Input LOX/HC Engine Data for SSTO Parallel Burn

FUEL	COOLANT	% H_2	MR	Vac Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
RP-1	HC	.00	2.42	312.0	28.7	10.3	3199	3160	4.81
CH_4	HC	.00	3.05	350.0	48.2	20.7	3108	2850	3.83
C_3H_8 - NBP	HC	.00	2.62	330.2	45.5	18.6	3148	2947	4.07
C_3H_8 - SC	HC	.00	2.70	332.8	49.0	20.7	3111	2617	3.88
RP-1	H_2	1.01	2.82	335.4	48.4	20.7	3090	2578	3.71
CH_4	H_2	1.09	3.53	359.8	48.2	20.7	3087	2746	3.69
C_3H_8 - NBP	H_2	1.04	3.13	344.9	49.2	20.7	3095	2683	3.77
C_3H_8 - SC	H_2	1.01	3.13	343.9	49.0	20.7	3094	2563	3.76

Table 3.4-6 Input LOX/HC Engine Data for SSTO Series Burn

FUEL	COOLANT	% H_2	MR	Vac Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
RP-1	HC	.00	2.42	312.0	28.7	10.3	5326	5739	8.0
CH_4	HC	.00	3.05	350.0	48.2	20.7	5181	5098	6.4
C_3H_8 - NBP	HC	.00	2.62	330.2	45.5	18.6	5239	5231	6.8
C_3H_8 - SC	HC	.00	2.70	332.8	49.0	20.7	5185	4682	6.5
RP-1	H_2	1.01	2.82	335.4	48.4	20.7	5129	4547	6.2
CH_4	H_2	1.09	3.53	359.8	48.2	20.7	5124	4837	6.1
C_3H_8 - NBP	H_2	1.04	3.13	344.9	49.2	20.7	5138	4728	6.3
C_3H_8 - SC	H_2	1.01	3.13	343.9	49.0	20.7	5137	4515	6.2

Table 3.4-7 Input LOX/HC Engine Data for UFRCV Trades.

Fuel	Coolant	% H_2	MR	Vac Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
RP-1	HC	.00	2.42	312.0	28.7	10.3	3336	3394	5.0
CH_4	HC	.00	3.05	350.0	48.2	20.7	3336	3132	4.1
C_3H_8 - NBP	HC	.00	2.62	330.2	45.5	18.6	3336	3190	4.3
C_3H_8 - SC	HC	.00	2.70	332.8	49.0	20.7	3336	2873	4.2
RP-1	H_2	.94	2.82	335.4	48.4	20.7	3336	2841	4.0
CH_4	H_2	1.01	3.53	359.8	48.2	20.7	3336	3029	4.0
C_3H_8 - NBP	H_2	.96	3.13	344.9	49.2	20.7	3336	2950	4.1
C_3H_8 - SC	H_2	.96	3.13	343.9	49.0	20.7	3336	2820	4.1

3.4.3.3 Procedures

The analysis procedures used to determine the reference vehicles and conduct the hydrocarbon engine trades are the same. The major task is to construct the vehicle sizing input file that incorporates the selected engine characteristics. Minor variations of this file are generated to span the sizing optimization parameter ranges of interest. These files are then processed by the vehicle sizing/performance procedures described in Section 3.2.1. The resulting vehicle output files provide the data to determine the minimum total vehicle dry mass configurations for each engine option.

Reference Vehicles

To determine an optimum SSTO for each LOX/LH2 mixture ratio engine an input parameter called the boost phase propellant mass fraction was manipulated; this parameter determines the boost phase duration. This parameter is the ratio between total boost phase propellant mass and total vehicle mass. This fraction directly determines the total propellant mass burned during the boost phase. The fraction was varied from .2 to .4 in .1 increments. This narrow range is justified by the SSTO's lack of sensitivity to this parameter as reported in the results.

For the UFRCV it was necessary to vary both the thrust ratio and boost duration optimization parameters for each LOX/LH2 engine of different mixture ratio used in the booster. Thrust ratio, defined as the ratio of total sea level thrust for the second stage to the total sea level thrust of the first stage, was varied from .1 to .4 in .1 increments. The boost duration was varied by changing the booster ideal velocity fraction, which is the fraction of total vehicle ideal velocity to be provided by the booster. The vehicle ideal velocity is the sum of the required orbital velocity and the velocity losses due to gravity, drag and nozzle pressure differences The booster ideal velocity fraction parameter was varied from .3 to .75 in .05 increments.

Trades

As noted above, the boost duration for SSTO vehicles is altered by varying the boost phase propellant mass fraction. This fraction was varied from .3 to .8 in .05 increments for both the series burn and parallel burn configurations. This range was altered for for the extreme ranges of thrust fraction values for the parallel burn configurations since some solutions do not exist in these extremities for all the boost phase propellant mass fraction values. The thrust fraction values were varied from .2 to .8 in .05 increments or until sufficient data was generated to obtain minimum total vehicle dry mass points.

When the SSTO program is run with a given input boost phase propellant mass fraction and thrust fraction, the resulting output contains the value for the percentage of hydrocarbon engine propellant. This value represents the percentage of the <u>total vehicle propellant</u> that is expended by the hydrocarbon fueled engines during the boost phase. This value is a direct indication of boost phase duration for both burn types of the SSTO. For the series burn SSTO the hydrocarbon engines are the only engines operating during boost phase, so the percentage of hydrocarbon engine propellant is also the percentage of total vehicle propellant expended during boost. For the parallel burn SSTO the amount of boost phase propellant expended by the hydrocarbon engines is a function of both the boost phase propellant mass fraction and the thrust fraction values. These combine to determine how much of the total vehicle propellant is expended by the hydrocarbon engines and, thus, the percentage of hydrocarbon engine propellant. Typically, vehicle dry mass is plotted against the percentage of hydrocarbon engine propellant in order to establish the trends of vehicle dry mass versus boost phase duration. An alternate method is to directly plot the total vehicle dry mass against the boost phase propellant mass fraction values.

Sensitivities

The sensitivity analyses proceeded by using the optimum vehicle input files, varying the parameter of interest by a slight amount while keeping all other parameter values constant, and then re-sizing the configuration. This was continued until the range of the parameters was covered.

3.4.4 Discussion of Analysis Results and Conclusions

3.4.4.1 Reference Vehicles (LOX/LH2)

SSTO

Figure 3.4-1 shows the total vehicle dry mass change for varying mixture ratios in a LOX/LH2 engine and percentage of vehicle mass burned during the boost phase. This percentage is merely the boost phase propellant mass fraction, described in section 3.4.3.3, multiplied by 100. This figure indicates that the dry mass minimum occurs at mixture ratio 8 with a boost phase propellant mass fraction of .2. Note, however that there is little difference between vehicles with mixture ratio 8 and 7, a negligible 0.1%. Also, dry masses vary little over the 20% to 40% range propellent mass in boost phase for the different mixture ratios, with a maximum change of 2.5% from the minimum point.

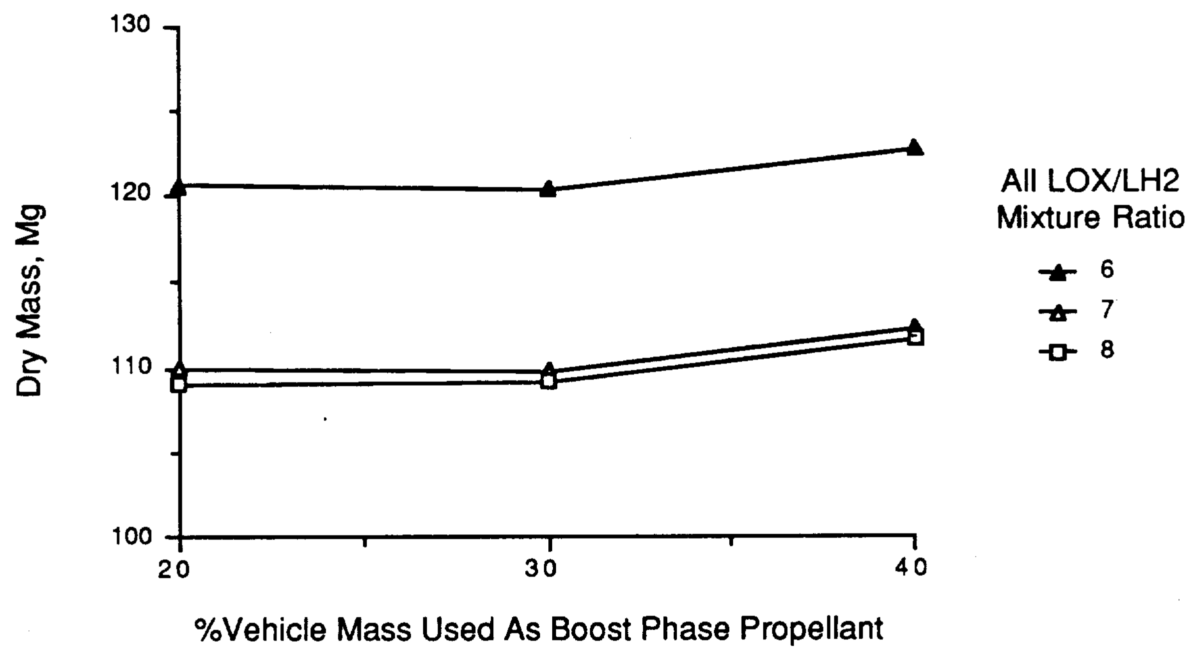


Figure 3.4-1 Total SSTO Dry Mass for LOX/LH2 Engine Mixture Ratio and Percentage of Vehicle Mass Expended During Boost Phase

The optimum all-hydrogen SSTO vehicle selected from the above results (reference vehicle) is described in Figure 3.4-2. It uses a mixture ratio of 8.

CONFIG. / CHAR.	Reference Vehicle SSTO BOOST	SUSTAIN
PROPELLANT TYPE	LO2/LH2	LO2/LH2
AREA RATIO	41.6	150.0
VAC Isp (secs)	429.1	454.8
MIXTURE RATIO	8	
NO. OF ENGINES	6.7	
GLOM (Mg)	1039.8	
PROPELLANT MASS (Mg)	895.2	
DRY MASS (Mg)	109.0	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.861	
BURN TYPE	N/A	
LENGTH (m)	48.67	
WING SPAN (m)	32.12	

Figure 3.4-3 SSTO Reference (LOX/LH2)Vehicle Description

UFRCV

Figure 3.4-3 shows total vehicle dry mass for the LOX/LH2 hydrogen engine using a mixture ratio 6 in the booster over a range of thrust ratios and percentage ideal velocity in the boost stage, which is the boost ideal velocity fraction, described in section 3.3.4.3, multiplied by 100. Similar data was generated for engines using the mixture ratios 7 and 8 . Each set of data indicated that the optimum value for thrust fraction was .2. Figure 3.4-4 thus compares the total vehicle dry mass for vehicles using engines of the three different mixture ratios over the percentage of ideal velocity in the boost stage range with thrust ratio of .2. The optimum point is for a vehicle using an engine with a mixture ratio of 7 and with a booster ideal velocity fraction of .5. However, a vehicle using an engine with a mixture ratio of 6 has only slightly greater mass. The reference vehicle description for this point is described in Figure 3.4-5.

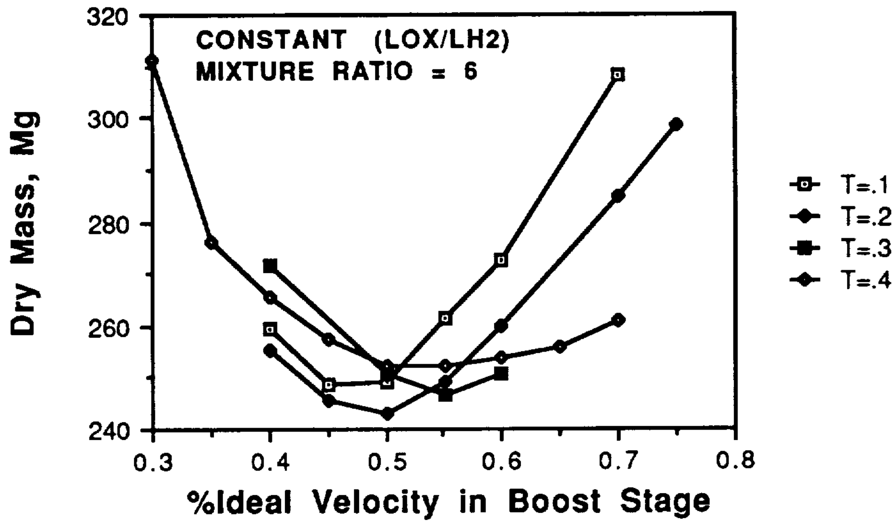


Figure 3.4-3 *Total UFRCV Dry Mass Versus Thrust Ratio and Booster Ideal Velocity Percentage for Mixture Ratio 6 LOX/LH2 Booster Engine*

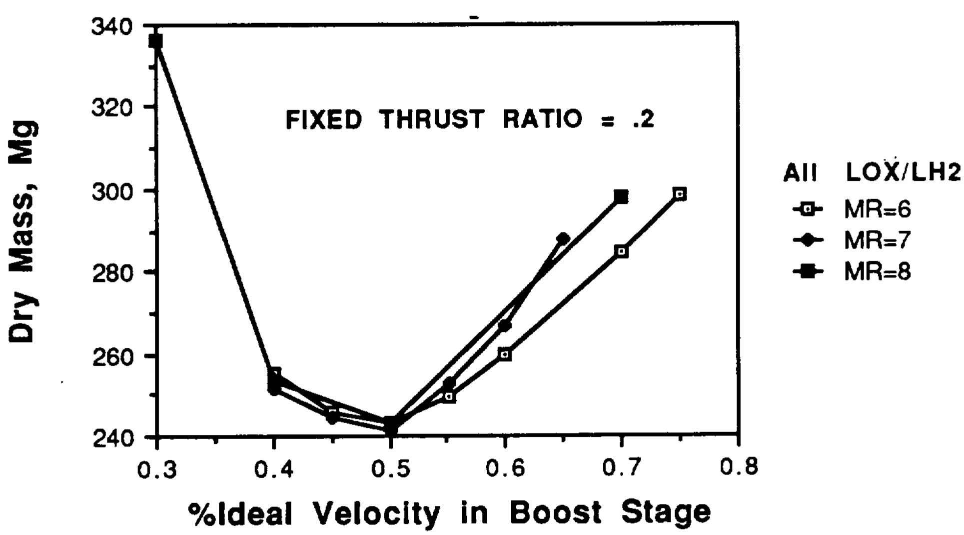


Figure 3.4-4 *Total UFRCV Dry Mass Versus Booster Ideal Velocity Percentage for Different Mixture Ratio Engines and Fixed Thrust Ratio*

CHAR. \ CONFIG.	UFRCV STAGE 1	STAGE 2
PROPELLANT TYPE	H/H	H/H
VAC Isp (sec)	439.9	463.6
VAC THRUST (MN)	18.2	4.0
MIXTURE RATIO	7.0	6.0
NO. OF ENGINES	4.8	1.6
STAGE MASS (Mg)	983	514
PROP. MASS (Mg)	807	415
DRY MASS (Mg)	150	91
MASS FRACTION	.82	.81
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	38.9	
LENGTH (m)	53.1	56.0
DIAMETER (m)	6.7	
GLOM (Mg)	152.7	

Figure 3.4-5 UFRCV Reference (LOX/LH2)Vehicle Description

3.4.4.2 Trade Studies

SSTO

Figure 3.4-6 shows the total vehicle dry mass change for the series burn vehicles over the range of percentage of hydrocarbon engine propellant, which is described in section 3.4.3.3 and is representative of the boost phase duration. Figures 3.4-7 and 8 show total vehicle dry mass plotted against the variable thrust fraction and percentage of hydrocarbon engine propellant for parallel burn vehicles. Selecting the minimum vehicle dry mass values for each shown in Figures 3.4-7 and 8, other curves are generated that show how total vehicle dry mass varies with respect to boost phase duration only, again using the percentage of hydrocarbon engine propellant . These curves are illustrated in Figure 3.4-9 and 10.

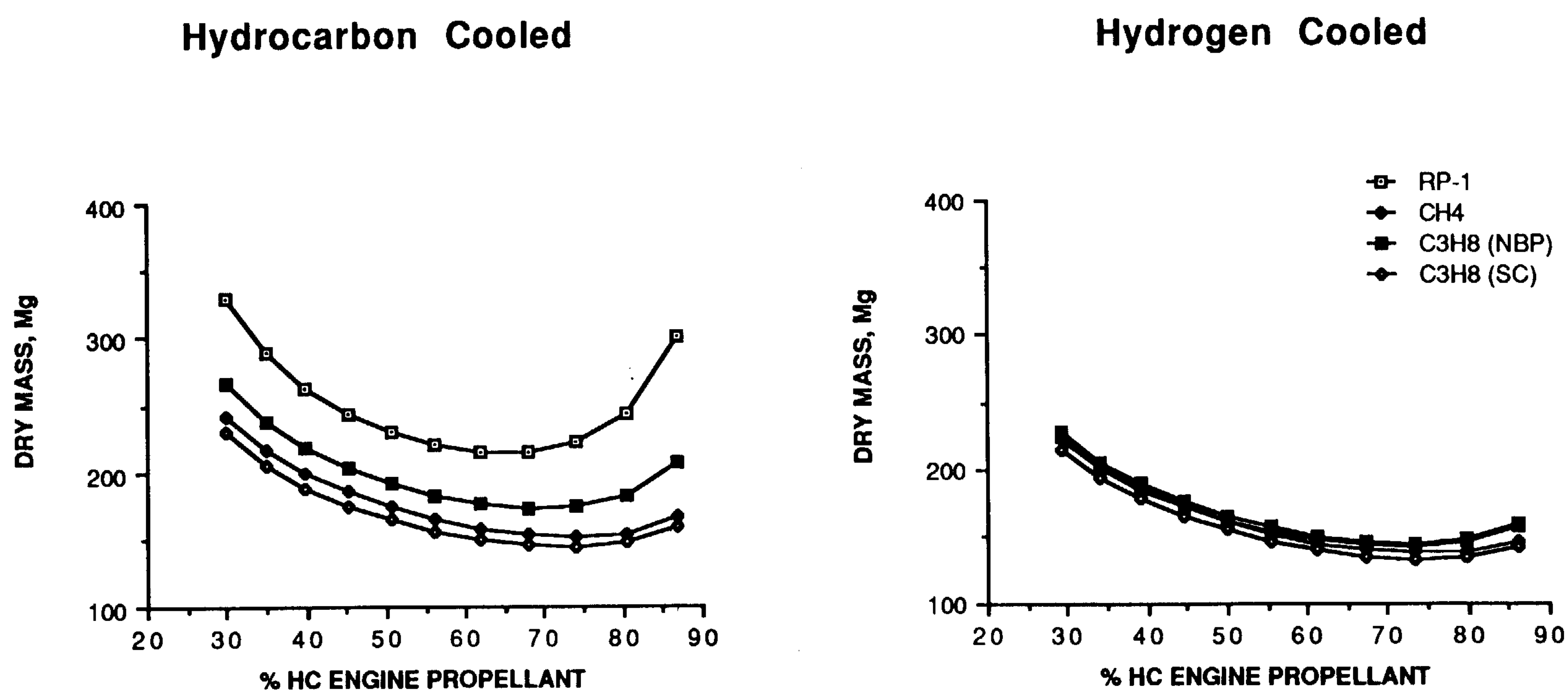


Figure 3.4-6 Total Vehicle Dry Mass for Series Burn SSTO's Using Hydrocarbon Engines

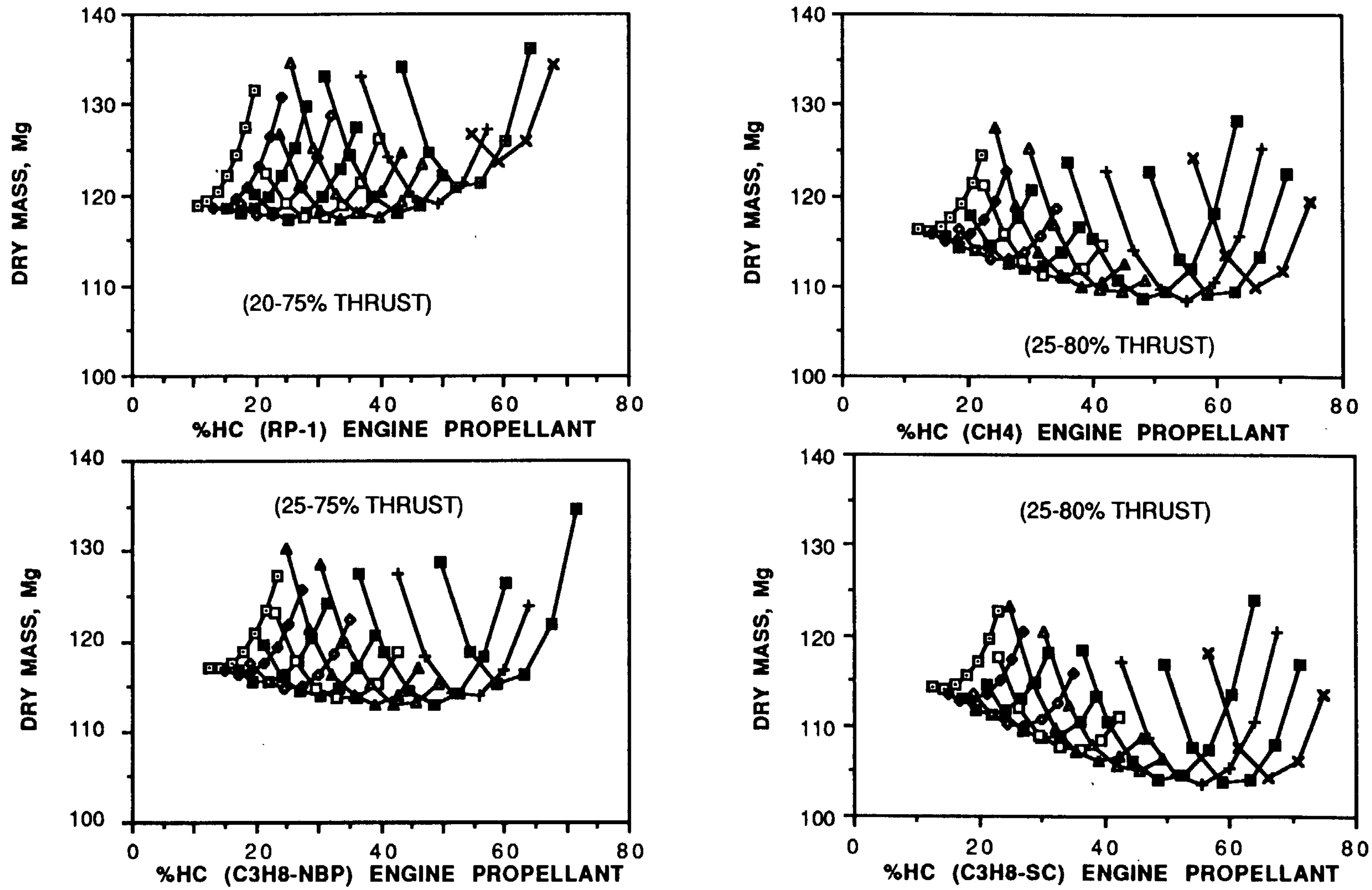


Figure 3.4-7 Total Vehicle Dry Mass for Parallel Burn SSTO's Using Hydrocarbon Engines with Fuel Cooling Showing Thrust Fraction Ranges

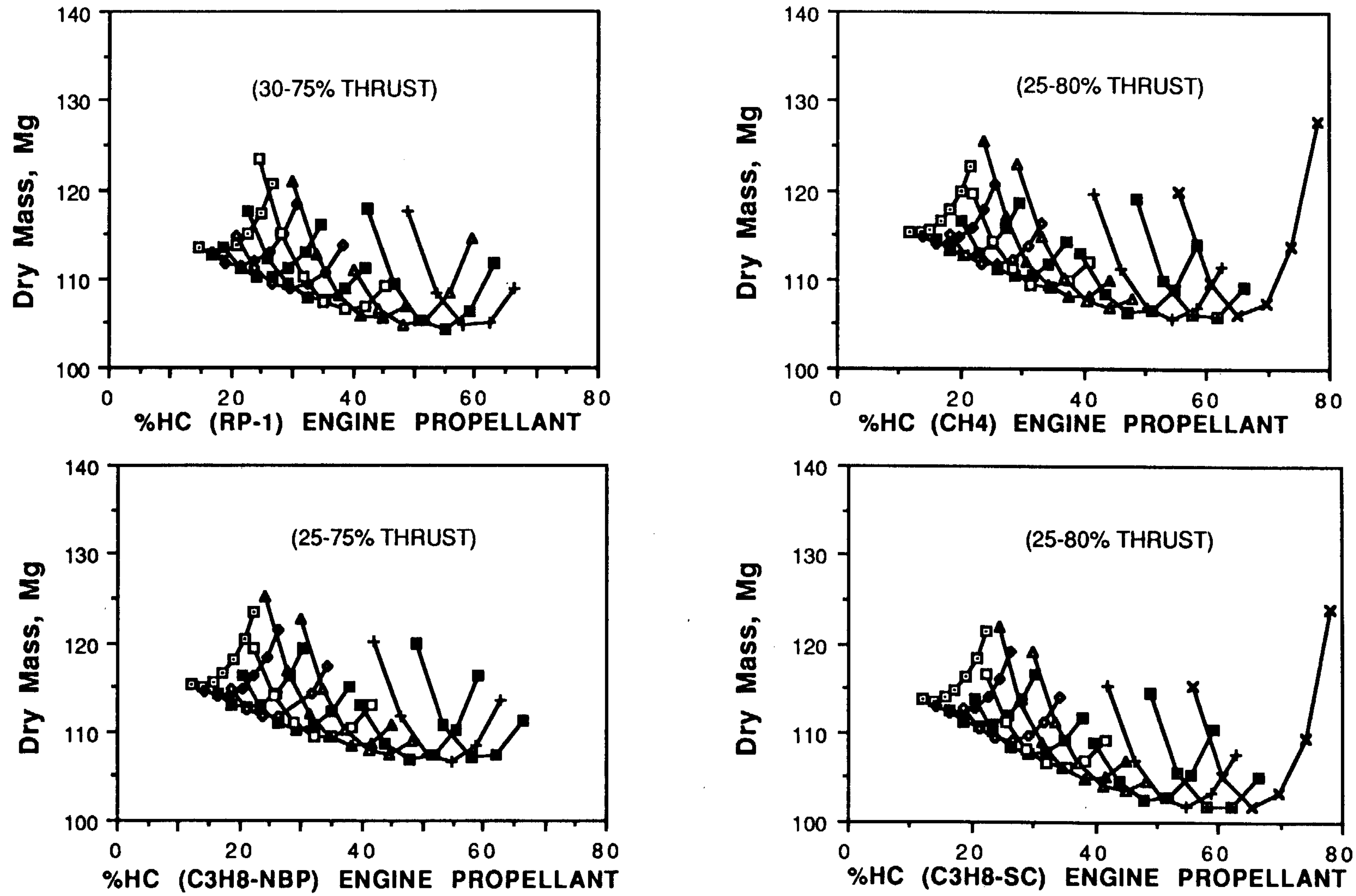


Figure 3.4-8 Total Vehicle Dry Mass for Parallel Burn SSTO's Using Hydrocarbon Engines with Hydrogen Cooling Showing Thrust Fraction Ranges

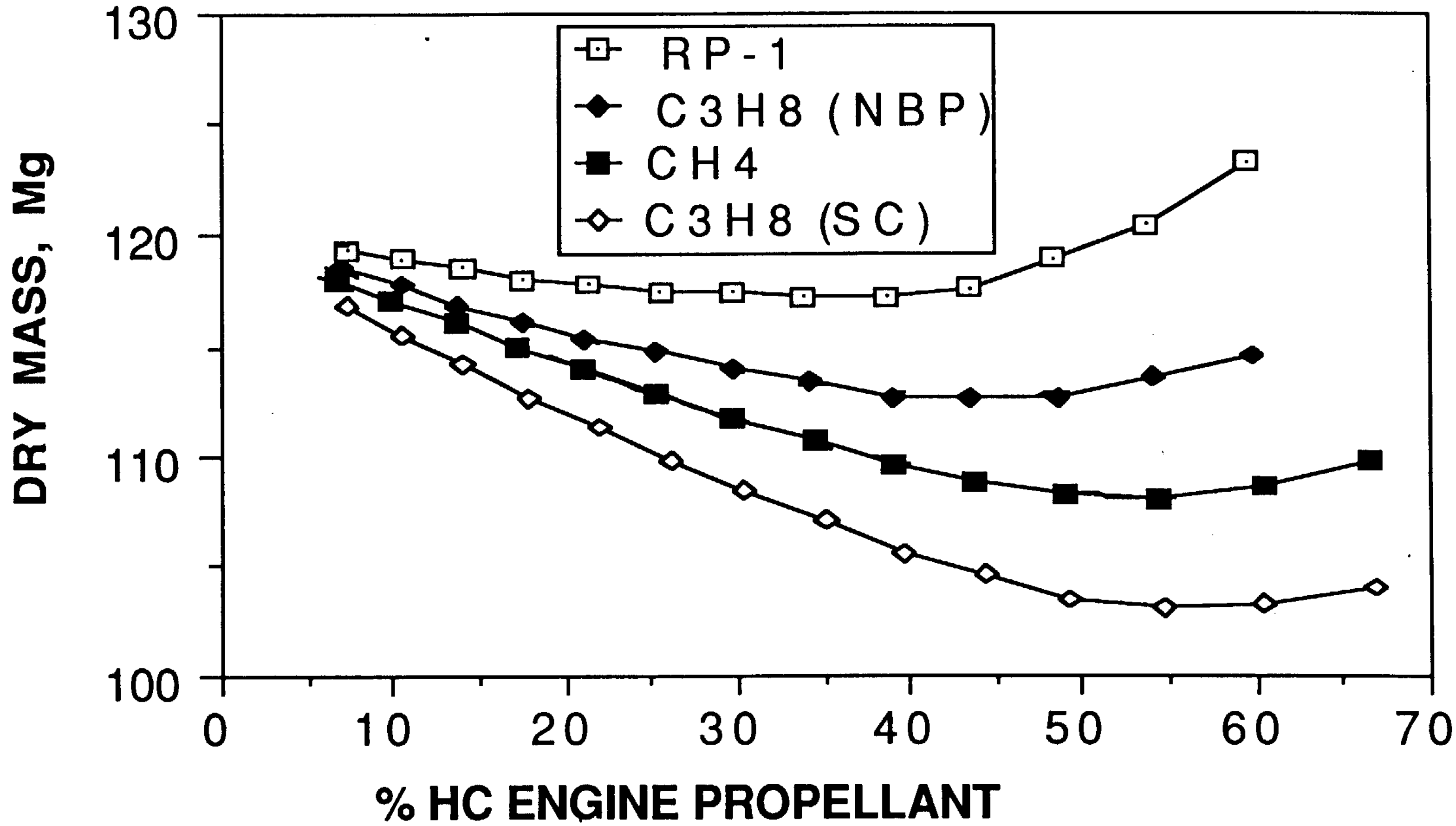


Figure 3.4-9 Total Vehicle Dry Mass for Parallel Burn SSTO's Using Hydrocarbon Engines with Fuel Cooling for Boost Duration Only

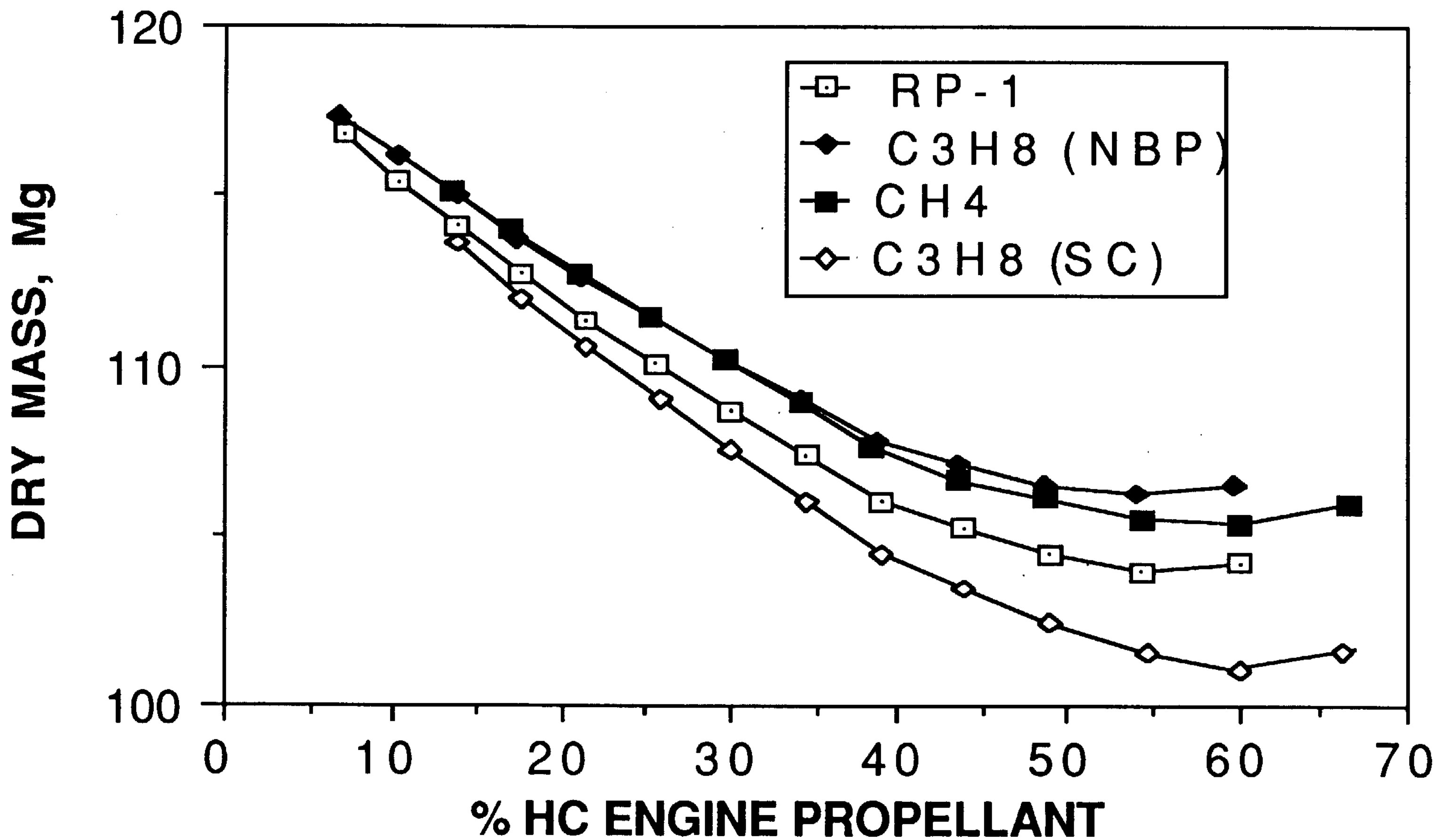


Figure 3.4-10 Total Vehicle Dry Mass for Parallel Burn SSTO's Using Hydrocarbon Engines with Hydrogen Cooling for Boost Duration Only

Figures 3.4-11 and 12 combine the series and minimum dry mass values, for each thrust fraction, of the parallel burn vehicles for hydrocarbon and hydrogen cooled configurations. Evaluating Figures 3.4-6 through 3.4-12 it can be seen that all parallel burn vehicles have a lower dry mass than the series burn vehicles. Furthermore, the use of subcooled propane always generates the lowest dry mass vehicle for either parallel or series burn. Figure 3.4-6 (series burn vehicles) shows a range of dry mass for hydrocarbon cooled vehicles of 143800 kg to 214700 kg or a range of almost 50 percent from the minimum value. The hydrogen cooled, series burn vehicles vary by only 9 percent. Figure 3.4-9 (parallel burn/hydrocarbon cooled vehicles) shows a range of dry mass of 103600 kg to 117400 kg which represents a 13 percent variation in dry mass, while the hydrogen cooled vehicles (Figure 3.4-10) show only a 5 percent variation in dry mass.

The optimum configurations for the eight hydrocarbon options were determined from Figures 3.4-11 and 3.4-12 by selecting those points, for each fuel/coolant combination, that represented the lowest vehicle dry mass. These were selected entirely from the parallel burn vehicles due to their lower mass. The optimum hydrocarbon vehicles are described in Figures 3.4-13 through 3.4-16; more detail exists in Appendix A.

A comparison of the optimum hydrocarbon vehicles' total dry mass and propulsion system mass to those of the reference, all LOX/LH2, vehicle is shown in Figure 3.4-17. The propulsion system mass includes the main engines, auxiliary propulsion elements, and the feed and pressurization subsystems. It does not include tankage. The vertical scales for both figures are percentage variation from the reference vehicle value. This percentage variation is calculated by subtracting the reference vehicle value from the value for the configuration of interest then dividing the result by the reference vehicle value and multiplying by 100 to obtain the percentage. Thus a +10% value indicates that the vehicle has a mass 10% greater than the reference vehicle value while a -10% indicates a value 10% less than the reference vehicle. Figure 3.4-17 shows that most hydrocarbon fueled vehicles still have lower dry masses than the optimized reference vehicle with the exception of hydrocarbon cooled RP-1 and NBP propane vehicles.

Optimum vehicles obtained, and their comparisons, showed five major trends: (1) large variations in dry mass between the four fuels were seen for hydrocarbon cooled candidates while hydrogen cooled candidates had maximum variations of 9 and 5 percent (between maximum and minimum dry mass vehicles) for series and parallel configurations respectively, (2) the use of sub-cooled propane generates the lowest total vehicle dry mass, (3) all parallel burn vehicles had lower total vehicle dry mass than any series burn vehicle, (4) the hydrogen cooled engines generated vehicles with lower total vehicle dry mass than their fuel cooled counter parts, and 5) most of the optimum parallel burn configurations have a lower mass than the reference vehicle, the exceptions being the RP-1 and NBP propane fueled engines with fuel cooling.

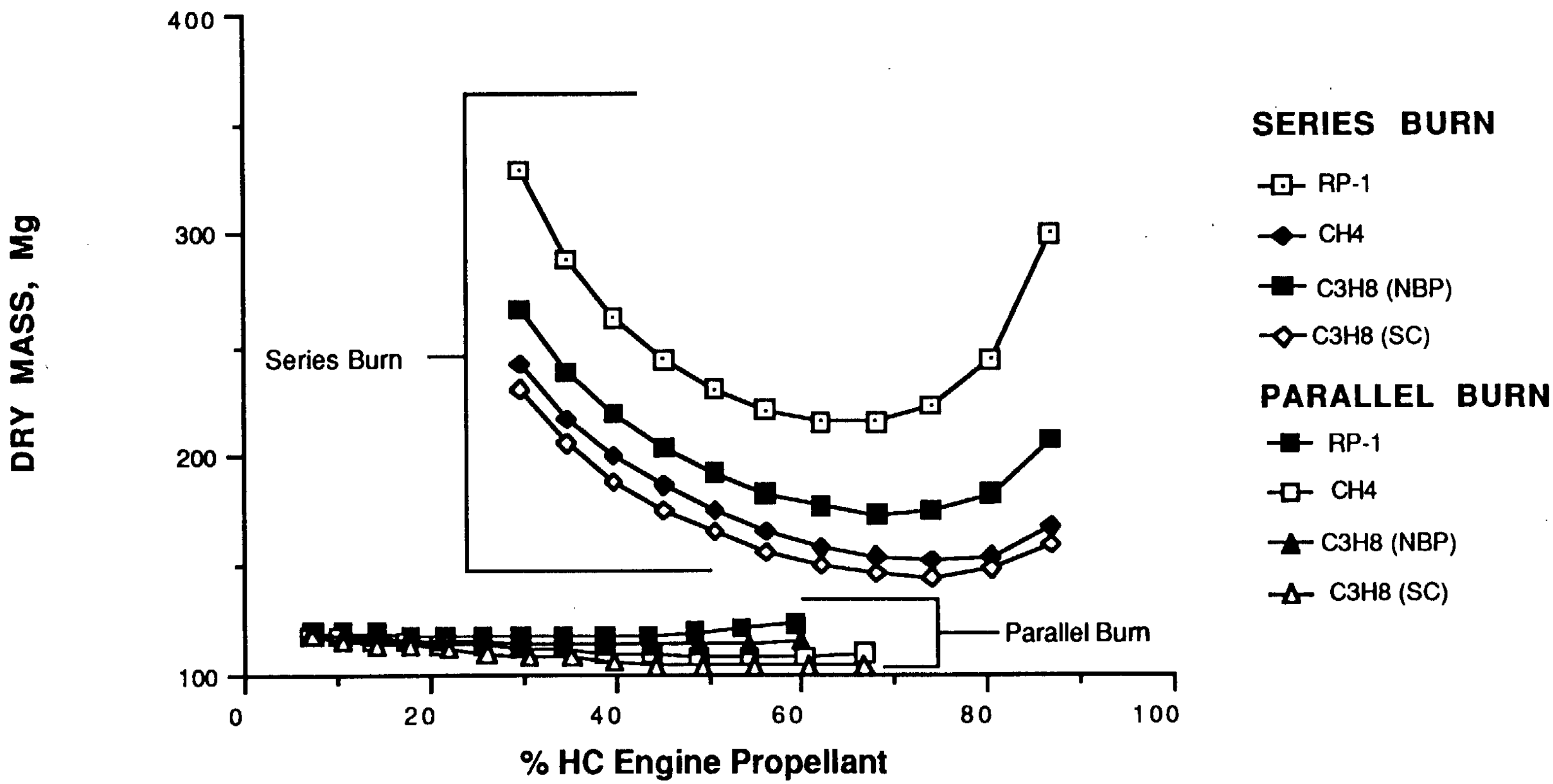


Figure 3.4-11 Series and Parallel Burn SSTO Total Vehicle Mass Versus Boost Phase Duration - Fuel Cooled Engines

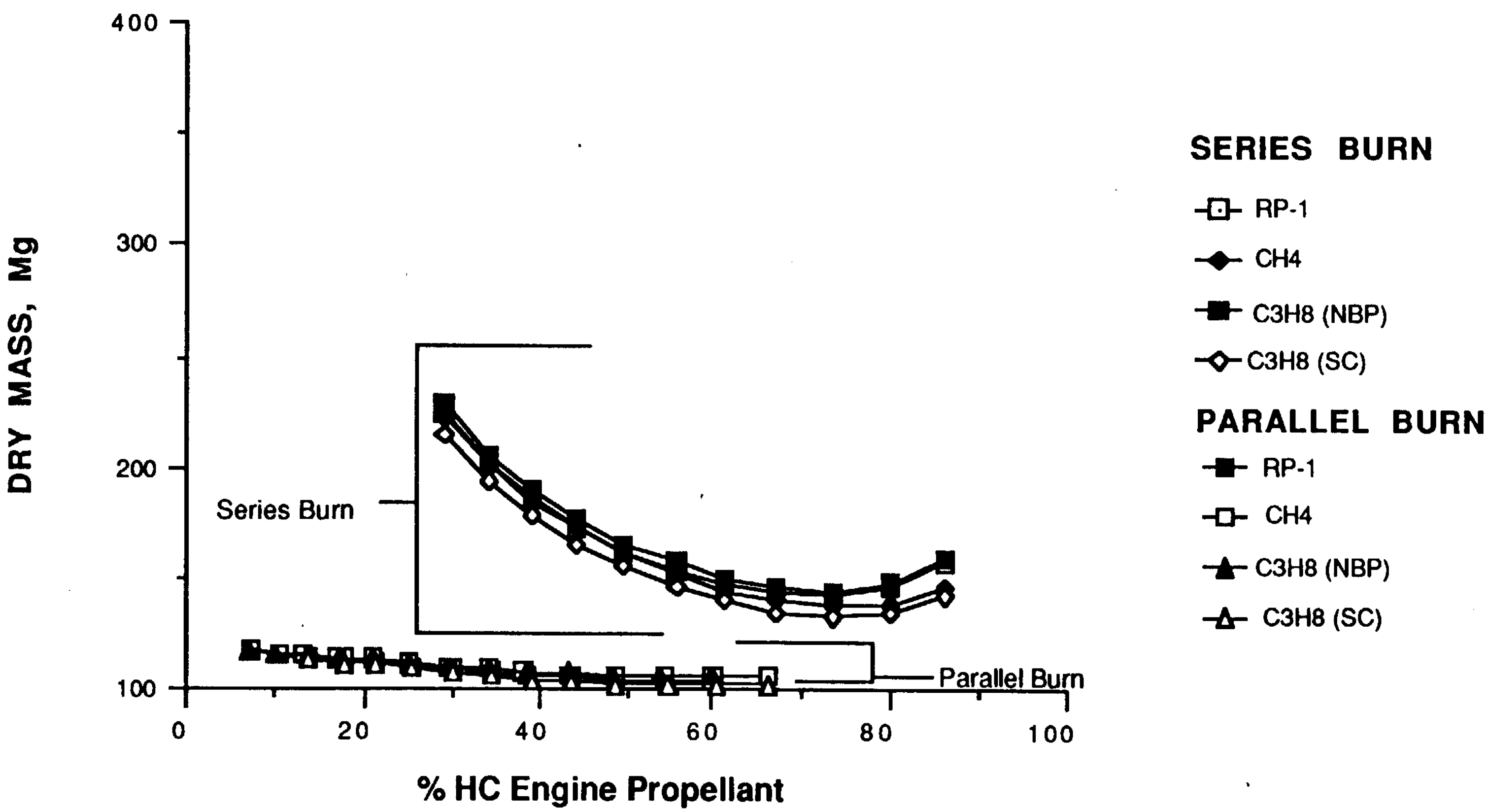


Figure 3.4-12 Series and Parallel Burn SSTO Total Vehicle Mass Versus Boost Phase Duration - Hydrogen Cooled Engines

CONFIG. / CHAR.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/RP1/RP1	LO2/LH2
AREA RATIO	28.7	41.6/150.0
VAC Isp (secs)	312.0	445.6/466.4
MIXTURE RATIO	2.42	8.0
NO. OF ENGINES	2.8	4.0
GLOM (Mg)	1227.5	
PROPELLANT MASS (Mg)	1071.9	
DRY MASS (Mg)	117.4	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.873	
BURN TYPE	PARALLEL	
LENGTH (m)	50.07	
WING SPAN (m)	33.05	

CONFIG. / CHAR.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/SP/SP	LO2/LH2
AREA RATIO	49.0	41.6/150.0
VAC Isp (secs)	332.8	445.6/466.4
MIXTURE RATIO	2.70	8.0
NO. OF ENGINES	3.9	2.4
GLOM (Mg)	1204.5	
PROPELLANT MASS (Mg)	1064.0	
DRY MASS (Mg)	103.6	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.883	
BURN TYPE	PARALLEL	
LENGTH (m)	47.35	
WING SPAN (m)	31.25	

Figure 3.4-13 Optimum SSTO Configuration Descriptions for RP-1/RP-1 and SP/SP

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/NP/NP	LO2/LH2
AREA RATIO	45.5	41.6/150.0
VAC Isp (secs)	330.2	445.6/466.4
MIXTURE RATIO	2.62	8.0
NO. OF ENGINES	3.4	3.2
GLOM (Mg)	1220.2	
PROPELLANT MASS (Mg)	1069.1	
DRY MASS (Mg)	113.1	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.876	
BURN TYPE	PARALLEL	
LENGTH (m)	49.39	
WING SPAN (m)	32.60	

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/CH4/CH4	LO2/LH2
AREA RATIO	48.2	41.6/150.0
VAC Isp (secs)	350.0	445.6/466.4
MIXTURE RATIO	3.05	8.0
NO. OF ENGINES	4.0	2.4
GLOM (Mg)	1213.9	
PROPELLANT MASS (Mg)	1067.9	
DRY MASS (Mg)	108.5	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.880	
BURN TYPE	PARALLEL	
LENGTH (m)	48.30	
WING SPAN (m)	31.88	

Figure 3.4-14 Optimum SSTO Configuration Descriptions for NP/NP and M/M

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/RP1/LH2	LO2/LH2
AREA RATIO	48.4	41.6/150.0
VAC Isp (secs)	335.4	445.6/466.4
MIXTURE RATIO	2.82	8.0
NO. OF ENGINES	4.0	2.4
GLOM (Mg)	1206.6	
PROPELLANT MASS (Mg)	1065.2	
DRY MASS (Mg)	104.3	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.883	
BURN TYPE	PARALLEL	
LENGTH (m)	47.76	
WING SPAN (m)	31.52	

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/SP/LH2	LO2/LH2
AREA RATIO	49.0	41.6/150.0
VAC Isp (secs)	343.9	445.6/466.4
MIXTURE RATIO	3.13	8.0
NO. OF ENGINES	4.2	1.9
GLOM (Mg)	1191.7	
PROPELLANT MASS (Mg)	1053.0	
DRY MASS (Mg)	101.9	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.884	
BURN TYPE	PARALLEL	
LENGTH (m)	47.38	
WING SPAN (m)	31.27	

Figure 3.4-15 Optimum SSTO Configuration Descriptions for RP-1/H and SP/H

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/NP/LH2	LO2/LH2
AREA RATIO	49.2	41.6/150.0
VAC Isp (secs)	344.9	445.6/466.4
MIXTURE RATIO	3.13	8.0
NO. OF ENGINES	3.9	2.4
GLOM (Mg)	1205.8	
PROPELLANT MASS (Mg)	1062.1	
DRY MASS (Mg)	106.5	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.881	
BURN TYPE	PARALLEL	
LENGTH (m)	48.19	
WING SPAN (m)	31.80	

CHAR. \ CONFIG.	Hydrocarbon Vehicle SSTO Hydrocarbon	Hydrogen
PROPELLANT TYPE	LO2/CH4/LH2	LO2/LH2
AREA RATIO	48.2	41.6/150.0
VAC Isp (secs)	359.8	445.6/466.4
MIXTURE RATIO	3.53	8.0
NO. OF ENGINES	4.2	2.0
GLOM (Mg)	1207.9	
PROPELLANT MASS (Mg)	1064.8	
DRY MASS (Mg)	105.9	
PAYLOAD (Mg)	13.6	
MASS FRACTION	.882	
BURN TYPE	PARALLEL	
LENGTH (m)	47.94	
WING SPAN (m)	31.64	

Figure 3.4-16 Optimum SSTO Configuration Descriptions for NP/H and M/H

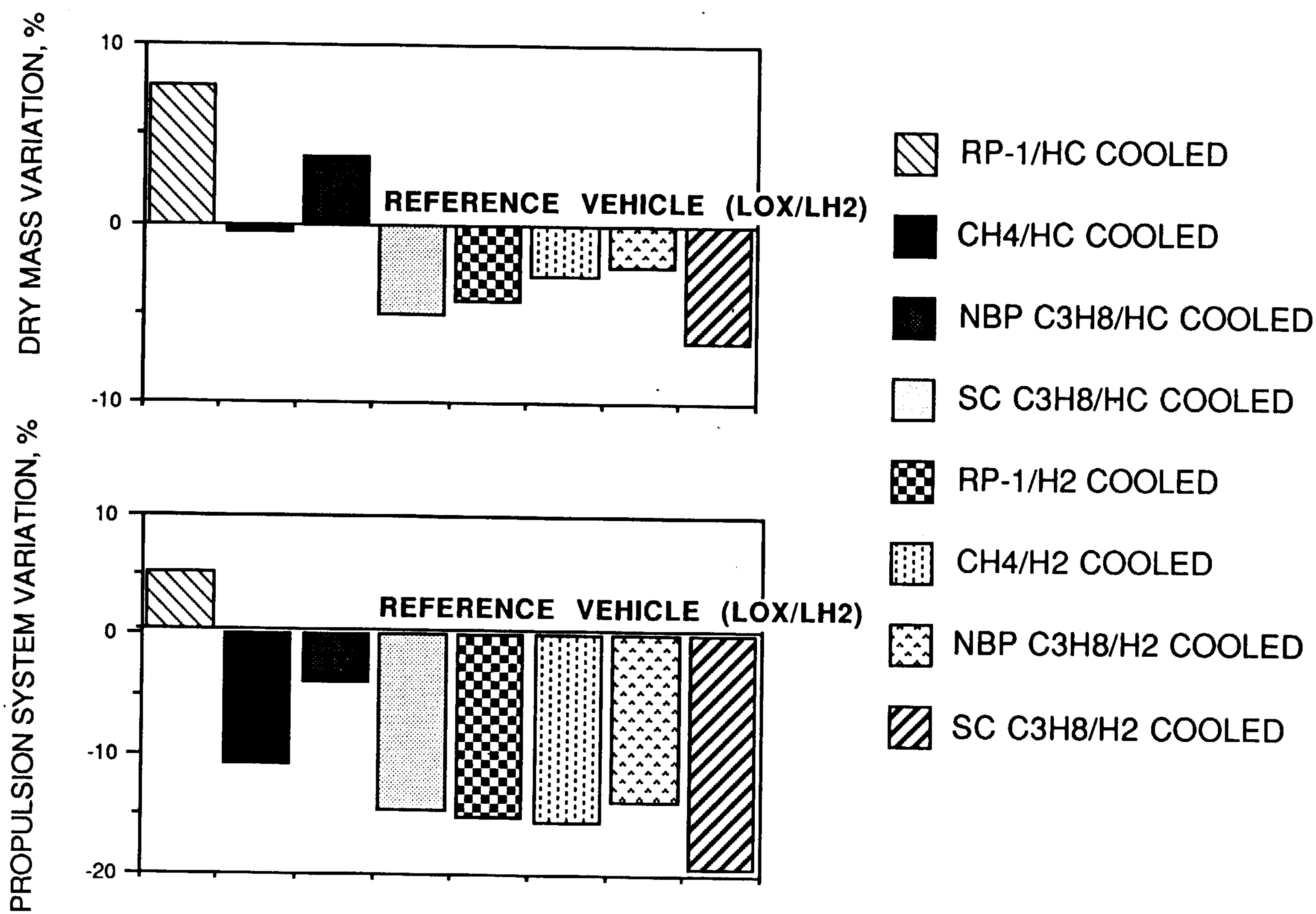


Figure 3.4-17 Comparison of Optimum SSTO Vehicles to Reference Vehicle (LOX/LH2)

UFRCV

Figures 3.4-18 and 3.4-19 show the change of total vehicle dry mass for varying hydrocarbon engine options, thrust ratio and percentage of total vehicle ideal velocity provided by the booster. For all the hydrocarbon engine options and thrust ratio values the minimum total vehicle dry mass fell between the 40% and 50% values for the percentage of total vehicle ideal velocity provided by the booster. At this percentage minimum point the thrust ratio of .2 usually generated the lowest mass vehicle with some exceptions. However, the differences between vehicle dry mass values for the different thrust ratio values, at this percentage minimum point, varied by little more than 1% from the absolute minimum. The optimum configurations for each hydrocarbon engine option were selected from this data in the same manner as for the SSTO described above. The optimum configurations are described in Figures 3.4-20 through 3.4-23 with more detail provided in Appendix A.

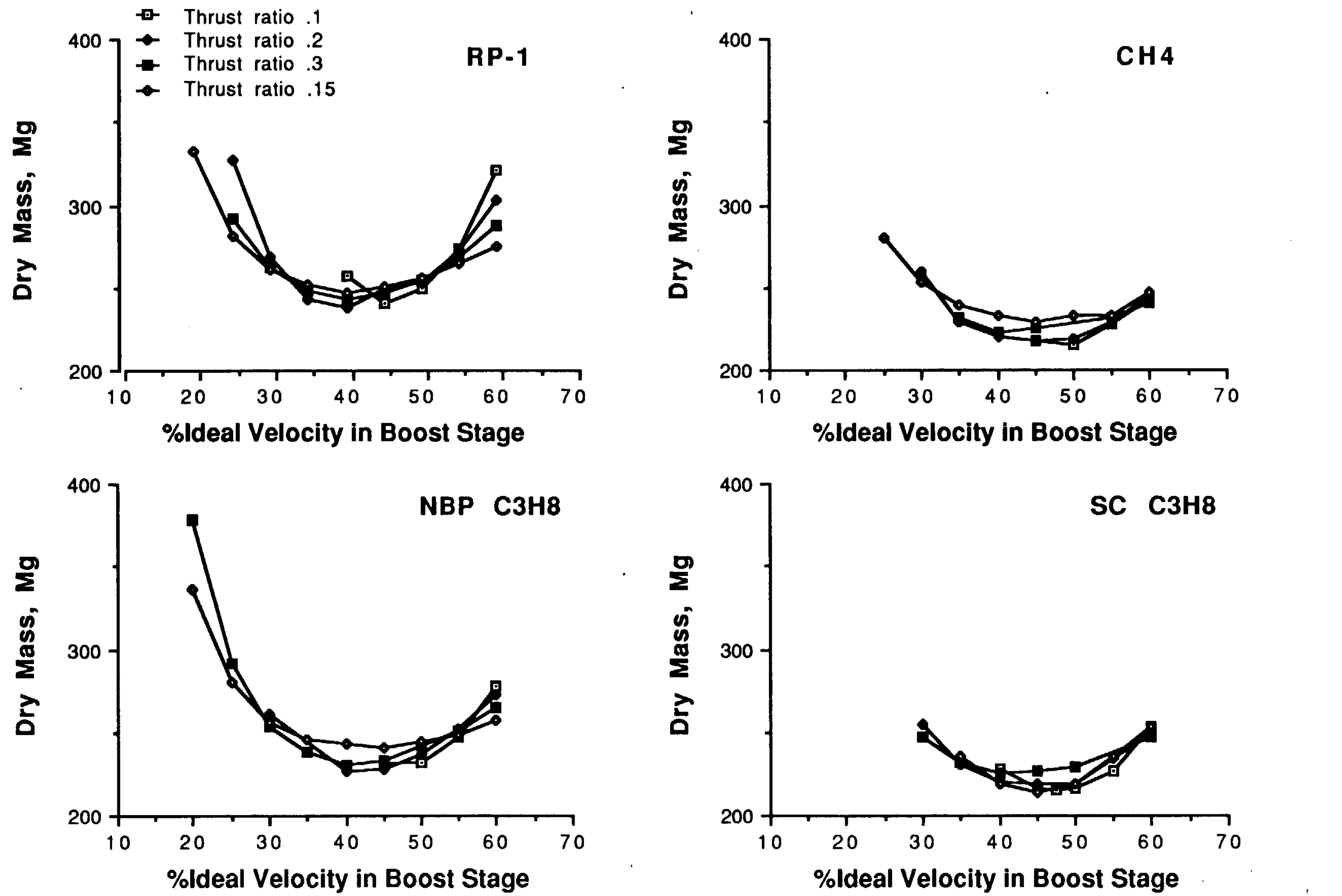


Figure 3.4-18 Total UFRCV Vehicle Dry Mass for Fuel Cooled Engines.

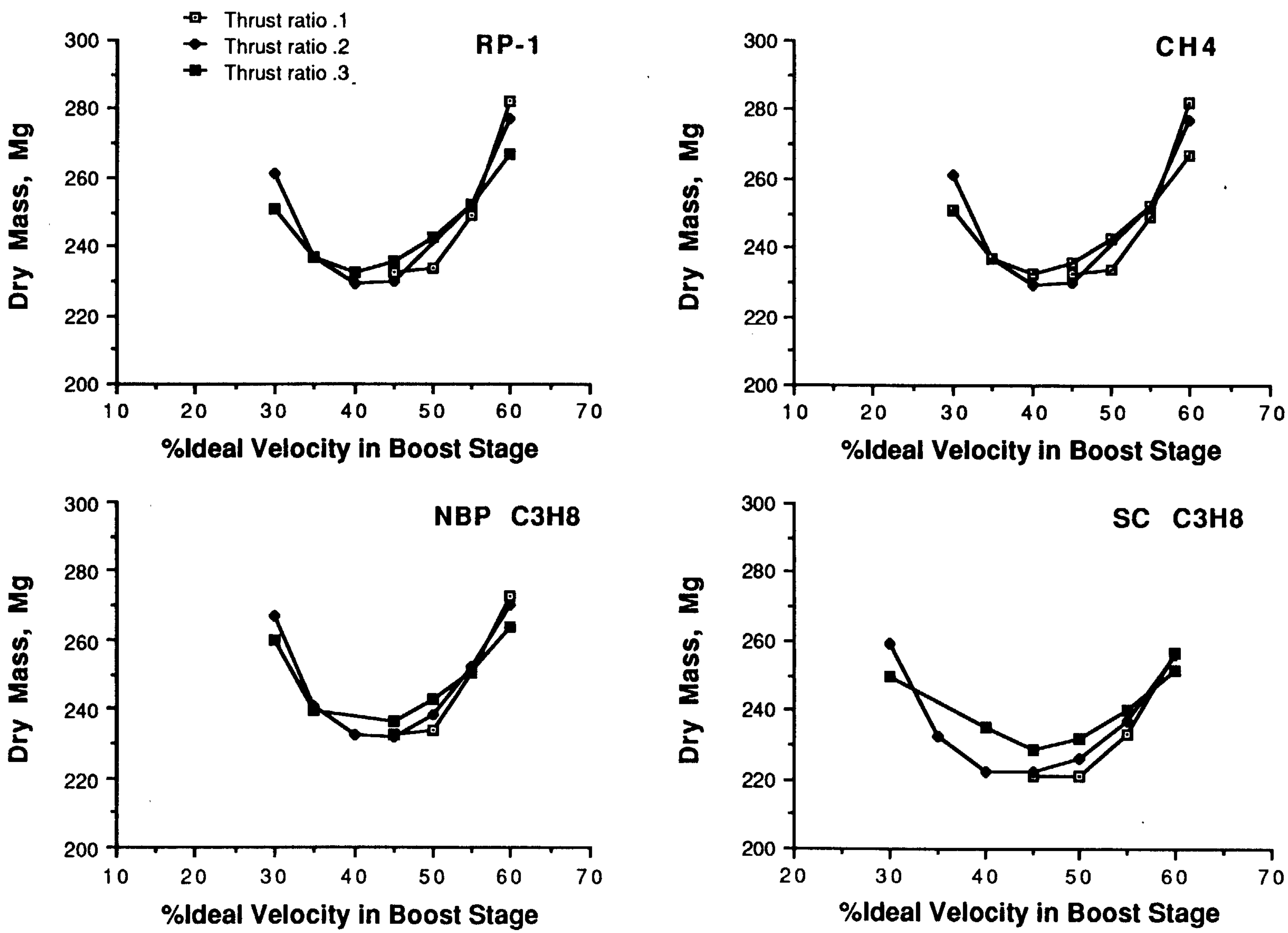


Figure 3.4-19 Total UFRCV Vehicle Dry Mass for Hydrogen Cooled Engines.

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	R/R	H/H
VAC Isp (sec)	312.0	463.6
VAC THRUST (MN)	25.1	5.3
MIXTURE RATIO	2.42	6.0
NO. OF ENGINES	3.2	2.1
STAGE MASS (Gg)	1.35	0.62
PROP. MASS (Gg)	1.17	.51
DRY MASS (Mg)	151	103
MASS FRACTION	.87	.82
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	37.5	
LENGTH (m)	45.3	58.8
DIAMETER (m)	5.4	
GLOM (Gg)	2.00	

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	M/M	H/H
VAC Isp (sec)	350.0	463.6
VAC THRUST (MN)	23.3	2.5
MIXTURE RATIO	3.05	6.0
NO. OF ENGINES	3.1	1.0
STAGE MASS (Gg)	1.35	0.38
PROP. MASS (Gg)	1.19	.30
DRY MASS (Mg)	140	76
MASS FRACTION	.88	.78
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	36.5	
LENGTH (m)	48.9	51.4
DIAMETER (m)	5.9	
GLOM (Gg)	1.76	

Figure 3.4-20 Optimum UFRCV Configuration Description - R/R and M/M

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	NP/NP	H/H
VAC Isp (sec)	330.2	463.6
VAC THRUST (MN)	21.9	4.7
MIXTURE RATIO	2.62	6.0
NO. OF ENGINES	2.9	1.9
STAGE MASS (Gg)	1.16	0.60
PROP. MASS (Gg)	1.01	.49
DRY MASS (Mg)	127	100
MASS FRACTION	.87	.82
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	34.3	
LENGTH (m)	45.4	58.6
DIAMETER (m)	5.4	
GLOM (Gg)	1.79	

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	SP/SP	H/H
VAC Isp (sec)	332.8	463.6
VAC THRUST (MN)	22.5	3.6
MIXTURE RATIO	2.70	6.0
NO. OF ENGINES	2.9	1.5
STAGE MASS (Gg)	1.26	0.48
PROP. MASS (Gg)	1.11	.39
DRY MASS (Mg)	127	88
MASS FRACTION	.88	.80
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	34.3	
LENGTH (m)	45.3	55.0
DIAMETER (m)	5.4	
GLOM (Gg)	1.77	

Figure 3.4-21 Optimum UFRCV Configuration Description - NP/NP and SP/SP

CONFIG. / CHAR.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	R/H	H/H
VAC Isp (sec)	335.4	463.6
VAC THRUST (MN)	21.7	4.7
MIXTURE RATIO	2.82	6.0
NO. OF ENGINES	2.9	1.9
STAGE MASS (Gg)	1.16	0.61
PROP. MASS (Gg)	1.00	.50
DRY MASS (Mg)	138	101
MASS FRACTION	.87	.82
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	35.1	
LENGTH (m)	48.1	58.7
DIAMETER (m)	5.7	
GLOM (Gg)	1.80	

CONFIG. / CHAR.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	M/H	H/H
VAC Isp (sec)	359.8	463.6
VAC THRUST (MN)	21.0	4.5
MIXTURE RATIO	3.53	6.0
NO. OF ENGINES	2.7	1.8
STAGE MASS (Gg)	1.08	0.61
PROP. MASS (Gg)	0.93	.50
DRY MASS (Mg)	136	101
MASS FRACTION	.85	.82
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	36.0	
LENGTH (m)	51.5	58.8
DIAMETER (m)	5.9	
GLOM (Gg)	1.73	

Figure 3.4-22 Optimum UFRCV Configuration Description - R/H and M/H

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	NP/H	H/H
VAC Isp (sec)	344.9	463.6
VAC THRUST (MN)	21.5	4.7
MIXTURE RATIO	3.13	6.0
NO. OF ENGINES	2.8	1.9
STAGE MASS (Gg)	1.13	0.61
PROP. MASS (Gg)	0.98	.50
DRY MASS (Mg)	137	101
MASS FRACTION	.86	.82
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	35.9	
LENGTH (m)	49.6	58.8
DIAMETER (m)	5.8	
GLOM (Gg)	1.78	

CHAR. \ CONFIG.	UFRCV STAGE 1	UFRCV STAGE 2
PROPELLANT TYPE	SP/H	H/H
VAC Isp (sec)	343.9	463.6
VAC THRUST (MN)	24.6	2.7
MIXTURE RATIO	3.13	6.0
NO. OF ENGINES	3.2	1.1
STAGE MASS (Gg)	1.45	0.38
PROP. MASS (Gg)	1.27	.30
DRY MASS (Mg)	153	77
MASS FRACTION	.88	.78
PAYLOAD (Mg)	29.5	
WINGSPAN (m).	37.7	
LENGTH (m)	49.3	51.5
DIAMETER (m)	5.9	
GLOM (Gg)	1.86	

Figure 3.4-23 Optimum UFRCV Configuration Description - NP/H and SP/H

Figures 3.4-24 through 3.4-26 provide a comparison between the eight configurations and the reference vehicle for total vehicle dry mass, booster engine package mass, and booster propulsion subsystem mass respectively. The latter subsystem includes the feed and pressurization systems only. The sum of engine package mass and propulsion subsystem mass is the total propulsion system mass, which does not include tankage or auxiliary propulsion. As for the SSTO, the comparisons are made on a percentage variation basis, which is calculated in the manner described above.

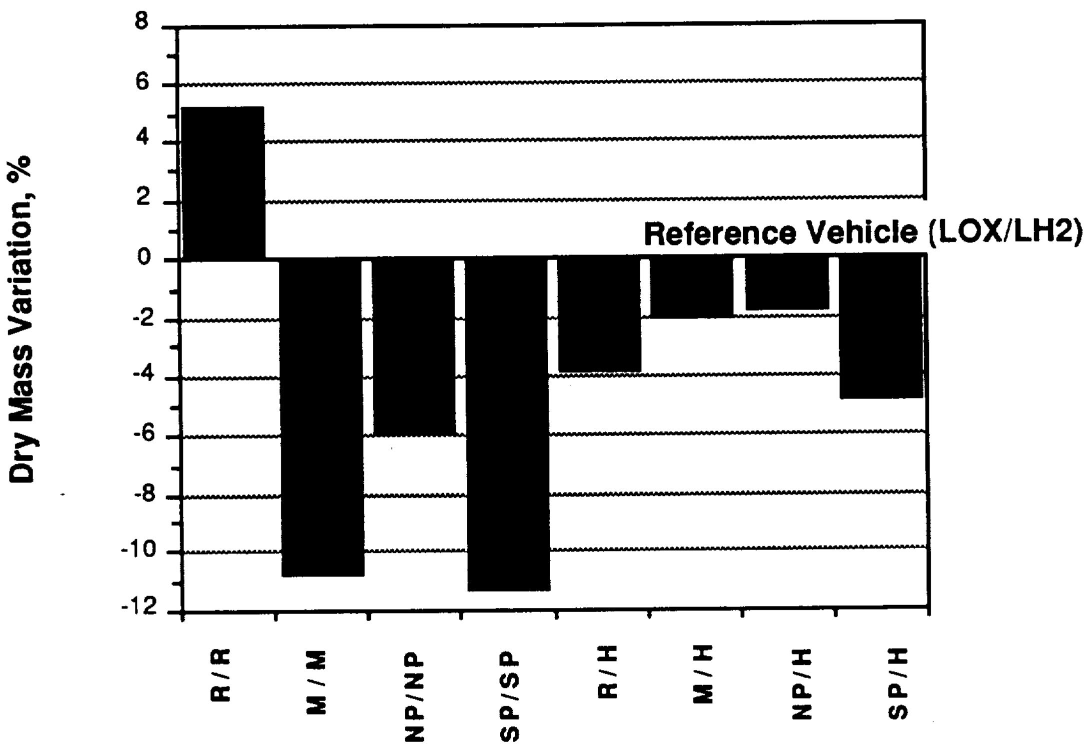


Figure 3.4-24 Comparison of Optimal UFRCV Configurations - Total Vehicle Dry Mass

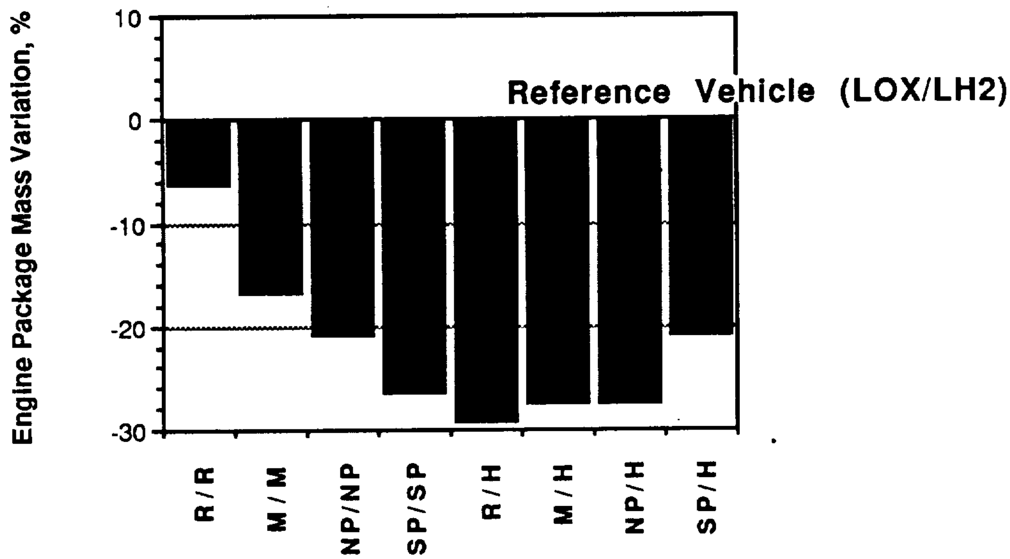


Figure 3.4-25 Comparison of Optimum UFRCV Configurations - Booster Engine Package Mass.

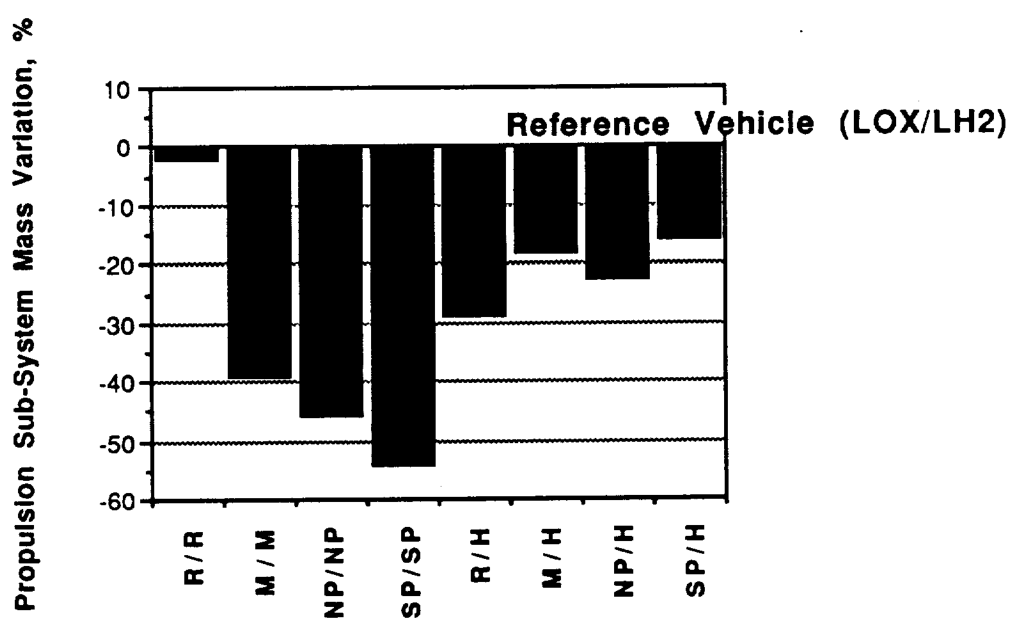


Figure 3.4-26 Comparison of Optimal UFRCV Configurations - Booster Propulsion Sub-system Mass.

These comparisons indicate some specific trends. All the hydrocarbon engine options generate vehicles with lower total dry mass than the reference configuration with the exception of the R/R option. This exception is due to the pressurization system penalty described in section 3.3.4.2. The subcooled propane engine using fuel cooling generates a vehicle with the lowest total dry mass. Unlike the SSTO analysis, the hydrogen cooled engine options were not quite as efficient as the fuel cooled engine options. This trend is believed due to the greater accuracy of the WASP model in tank sizing and the fact that the SSTO vehicle can store the hydrogen coolant with the hydrogen fuel used during the sustainer phase while a booster must have an added tank for the hydrogen coolant. This added tank is a penalty for using hydrogen cooling. None of the hydrogen cooled engine options generated vehicles significantly lower in mass than the reference, with the largest decrease being 5 percent while the majority of the reductions were near 3 percent.

The engine package mass trend is clear. The engines with the highest thrust to weight value, or lowest mass for thrust delivered, resulted in the greatest decrease in engine package mass from the reference case. The propulsion subsytem mass reduction from the reference case was substantial for all the options, except for the R/R option, but was roughly the same value for all of them.

3.4.4.3 Sensitivities

SSTO

Figures 3.4-27 and 3.8-28 show the parallel burn sensitivities, fuel and hydrogen cooled vehicles respectively, for the three sensitivity parameters. Vehicles tended to display a different sensitivity to specific impulse based upon the hydrocarbon fuel with fuel cooling. This may be due to the fact that the differently fueled optimum vehicles had different percentages of the hydrocarbon engine propellant factor. The sensitivity due to the engine thrust to weight ratio was almost identical for all the propellants (either fuel or hydrogen cooled). However, the sensitivity for fuel cooled vehicles tends to increase with increasing vehicle dry mass of the original, optimum configuration. Sensitivity to mixture ratio varied substantially between propellants. The trend was an inverse of the bulk density, with CH_4 by far the most sensitive, followed by NBP propane, SC propane, and RP-1. The hydrogen cooled engine options showed less sensitivity to the parameters than the fuel cooled options.

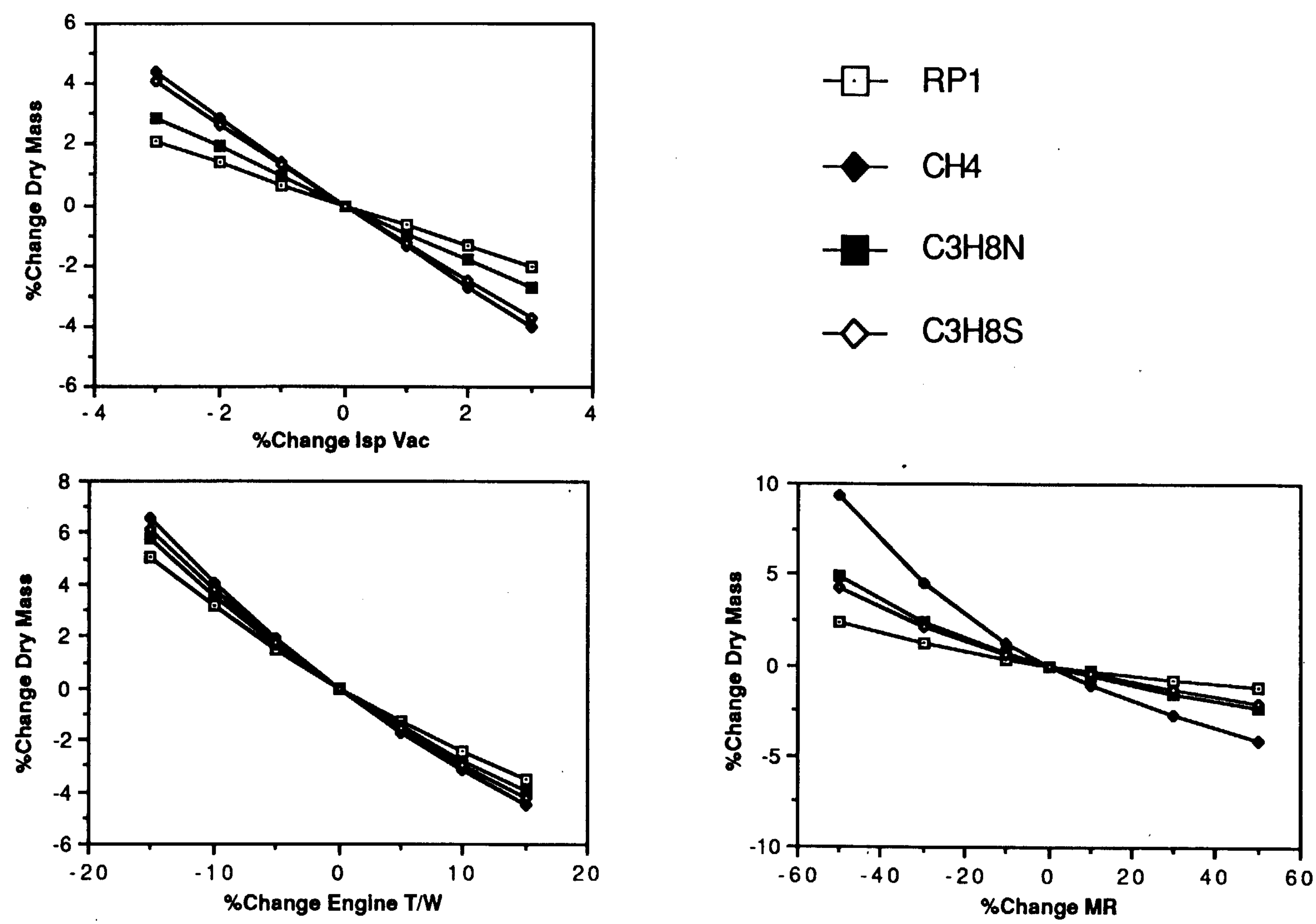


Figure 3.4-27 Sensitivities of Parallel Burn SSTO Optimum Configurations - Fuel Cooled Engines

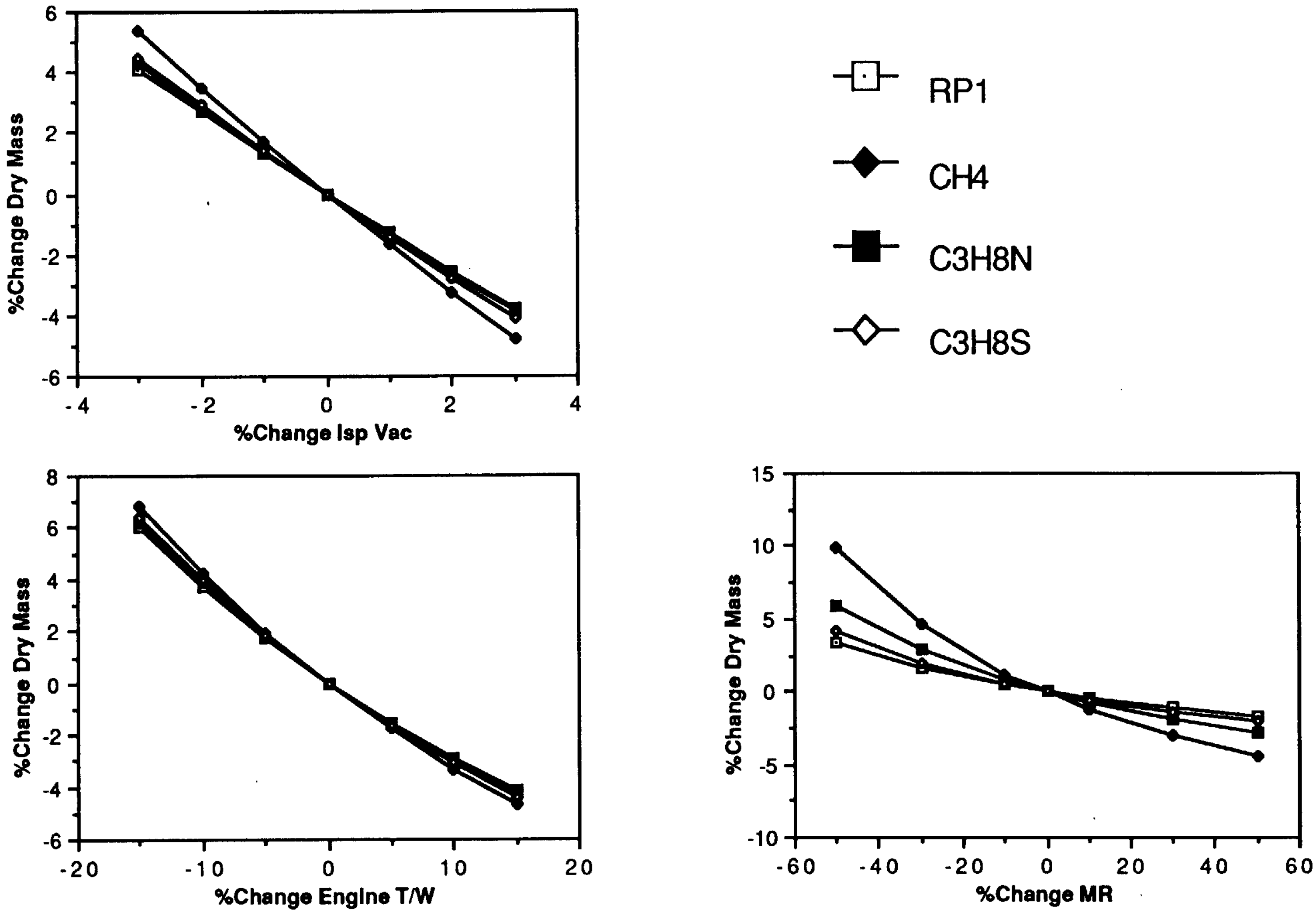


Figure 3.4-28 Sensitivities of Parallel Burn SSTO Optimum Configurations - Hydrogen Cooled Engines

UFRCV

Figures 3.4-29 and 30 show optimum vehicle sensitivity to specific impulse, engine thrust to weight and engine mixture ratio booster parameters for the fuel and hydrogen cooled engine options respectively.

Some graphs in figures 3.4-29 and 3.4-30 show minor discontinuities in the curves. These are due to problems in obtaining vehicle sizing solutions very near to the optimum vehicle design. The WASP program has built in discontinuities due to geometry constraints on the UFRCV. In particular, the requirement to space the forward and aft wings of the booster properly in order to fit the orbiter payload bay, and constraints on the frustum shaped nose of the booster generate discontinuities in the narrow range of vehicle total dry masses about a fixed vehicle design. This occurs most markedly when the perturbations in vehicle characteristics, as when a sensitivity analysis is conducted, generates vehicle total dry mass changes of less than 5%. In addition, the vehicle sizing attempts to satisfy numerous other constraints such as minimal structural mass, required payload/orbit performance and not exceeding maximum acceleration or dynamic pressure. Satisfaction of these constraints does not happen completely simultaneously and variations in when the parameters are satisfied can result in discontinuities near the optimum vehicle point. However, the general trends indicated in the sensitivity analysis results are believed correct.

The vehicle total dry mass does not exhibit significant sensitivity to either changes in booster engine mixture ratio or engine thrust to weight. Significant sensitivity is assumed to be a change in vehicle dry mass in excess of two percent for a small change in the parameter of interest. In contrast, the vehicle dry mass values show some sensitivity to specific impulse values. It is clear when comparing figure 3.4-29 and 30 that the hydrogen cooled engines show a greater sensitivity to mixture ratio than the fuel cooled options. This is due to the different tank arrangements used in vehicle sizing for the two coolant options. For the fuel cooled options the forward tank was oxidizer and the aft tank was fuel. For the hydrogen cooled options the arrangement was fuel in the forward tank and oxidizer in the aft tank and a hydrogen tank was added. This added hydrogen tank, combined with the geometry constraints on the UFRCV, increased sensitivity to engine mixture ratio.

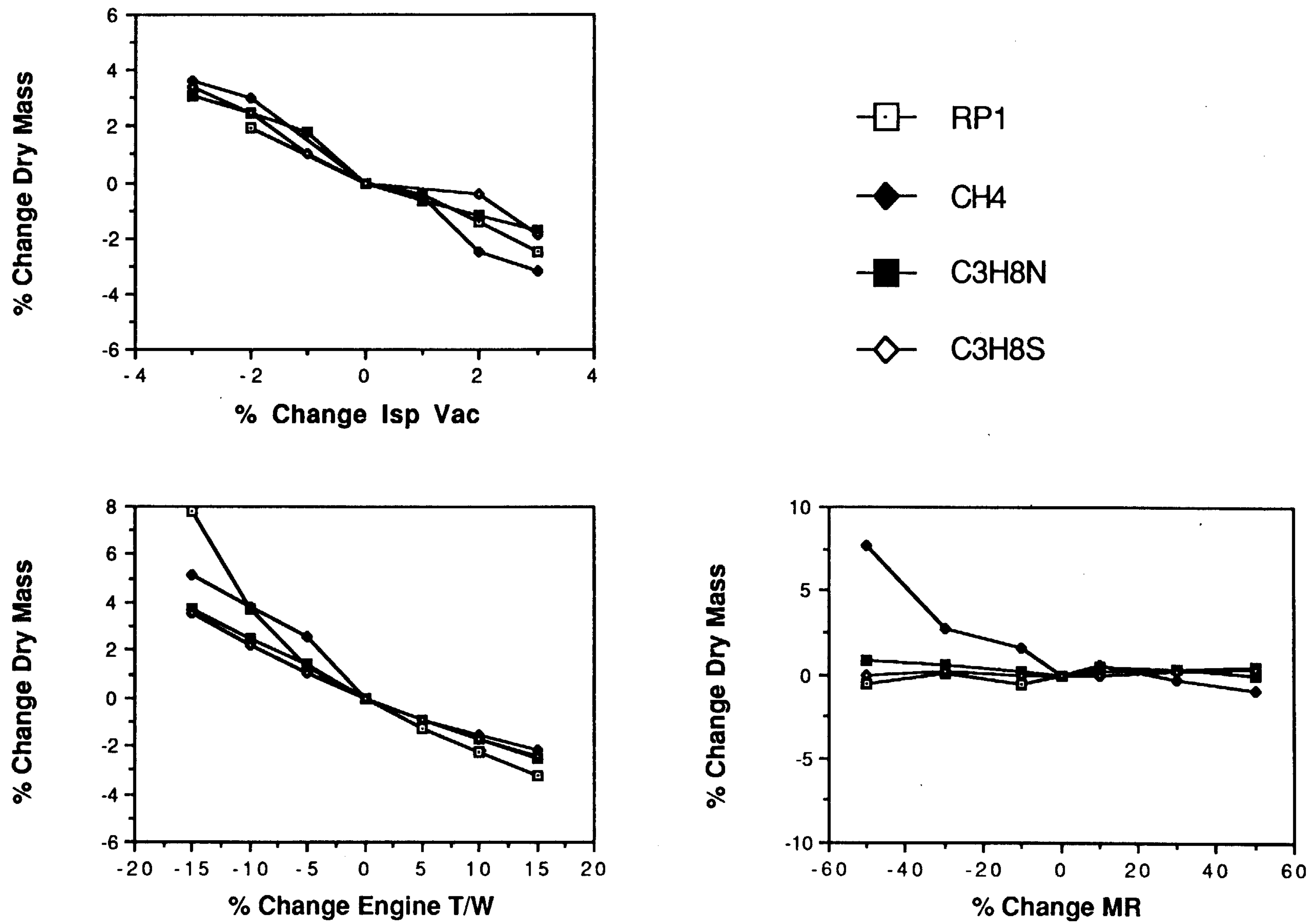


Figure 3.4-29 Sensitivities of Optimum UFRCV Configurations-Fuel Cooled Engines

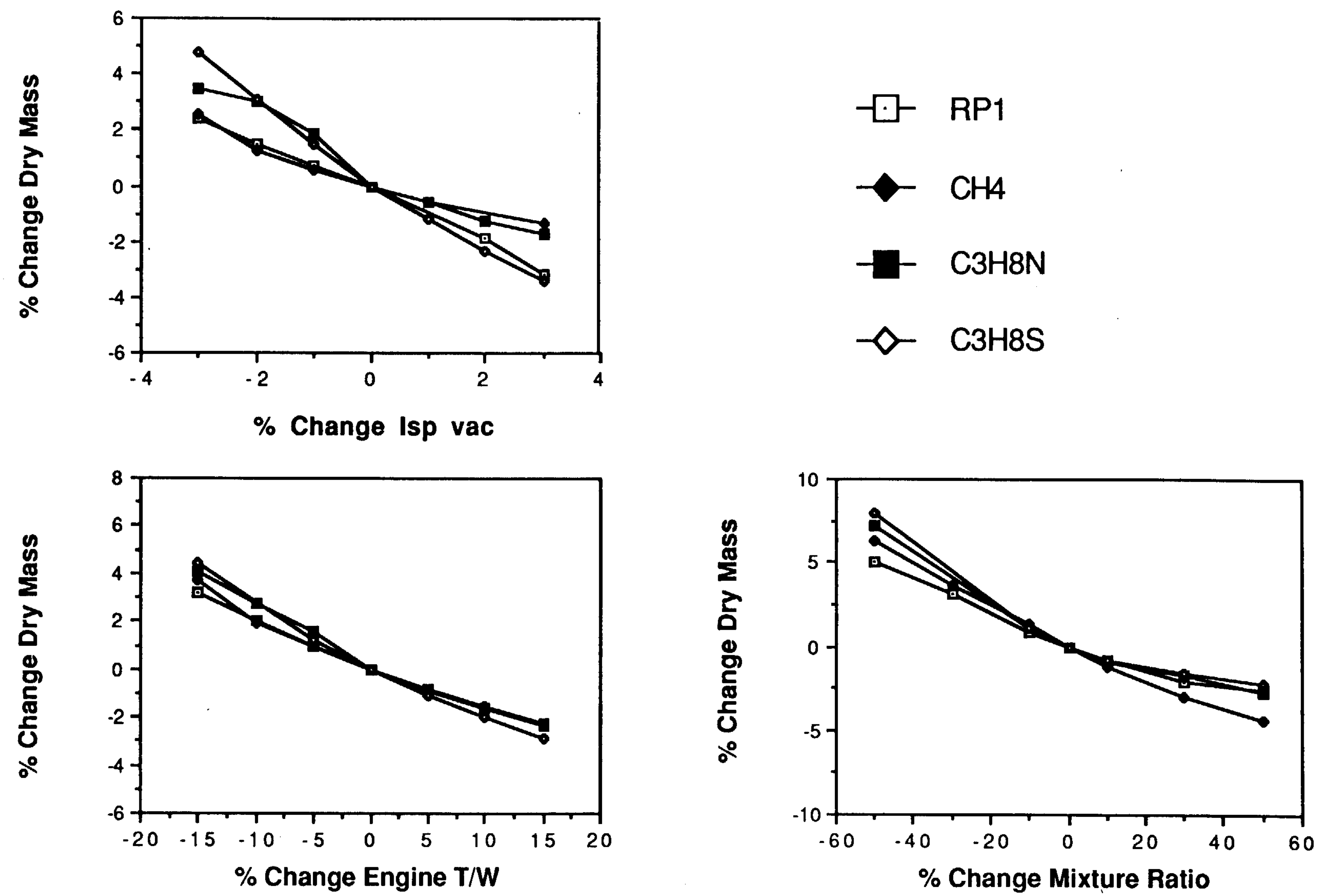


Figure 3.4-30 Sensitivities of Optimum UFRCV Configurations-Hydrogen Cooled Engines

3.5 Task 1.3 - Conduct Cross-Feed Analysis

3.5.1 Objective

The objective of this subtask is to determine the impact of cross feeding propellants from the booster to the second stage of the UFRCV when the booster is using hydrocarbon engines with hydrogen as a coolant.

3.5.2 Summary of Task Activity

The standard vehicle sizing analysis was conducted using the sizing ground rules and adjusting the input cases for use of cross feeding propellants. The major adjustments made were to increase the feed system weights in both stages. Optimization of the UFRCV for the four hydrocarbon engine options was conducted for the boost duration parameter and thrust ratio.

The optimum vehicles for the four hydrocarbon engine options were identified and compared to the results from Subtask 1.2.

3.5.3 Discussion of Analysis Procedure

3.5.3.1 Ground Rules and Assumptions

As in previous tasks, the standard sizing ground rules were used along with the UFRCV performance requirements and subsystem design parameters. The latter were adjusted to account for the use of cross feeding propellants. This adjustment involved altering the weight estimating relationships for the feed systems of both stages of the UFRCV. The relationships were altered to account for the additional feedlines in both stages as well as the increased size of the feedlines in the booster. The relationships were determined based upon the assumption that the crossfeed lines, from the booster to the orbiter, fed into the orbiter engines directly, rather than leading to the orbiter tanks. This assumptions limited the total length of the feed lines for both stages.

It was believed necessary to optimize on both boost duration (or staging velocity) and thrust ratio as was done in the trades analyses. What thrust ratio values would determine the minimum vehicle dry mass could not be anticipated due to the substantial change in vehicle design. Thus thrust ratio values in the range of .1 to .3 were examined.

3.5.3.2 Input Data

The input required for this task included the engine data, the baseline vehicle input file for sizing and the new weight estimating relationships for the feed systems. The engine data used was that shown in Table 3.5-7 for the hydrogen cooled engines.

3.5.3.3 Procedure

Six different input files were created for each hydrocarbon engine option and for each thrust ratio value for a total of 72 files. The six input files spanned a range of booster duration, defined as before as the fraction of total vehicle ideal velocity supplied by the booster, from .35 to .6 in .05 increments. These files were processed in the usual manner to determine the vehicle weights and geometries. Total vehicle dry weights for the 72 cases were plotted in the usual manner. The optimum cases, based on lowest total dry dry weight, were selected for each hydrocarbon engine option. These optimums were compared to the reference vehicle and the optimum vehicles found in the trade studies of Subtask 1.2.

3.5.4 Discussion of Analysis Results and Conclusions

The plots of total vehicle dry mass versus percentage of total vehicle ideal velocity for the different hydrocarbon fuels for different thrust ratios are shown in Figure 3.5-1. The vehicle optimums were selected on the basis of this figure. Figures 3.5-2 through 3.5-4 compare the total vehicle dry mass, and booster engine package and propulsion subsystem masses respectively for the four optimum cross-fed configurations to the reference vehicle and to the four optimum (non-crossfed) configurations using hydrocarbon engines with hydrogen coolant as described in Section 3.4.4. Figure 3.5-5 compares these optimum cross-fed configurations to the reference vehicle and the optimum configurations for hydrocarbon engines with fuel cooling on a total vehicle dry mass basis.

It is readily apparent from Figure 3.5-2 that the use of cross feeding propellants from the booster to the second stage markedly improves the weight reduction results for hydrogen cooled engines. The weight reduction, from the reference case, for cross fed configuration is between 11 and 13 percent. This implies that the use of cross feeding increases the weight reduction, found when cross feeding was not used, by 8 to 10%.

For both sets of vehicles, with cross feed and without, the engine thrust to weight ratios are identical since the same hydrogen cooled engines are used for both. However, as Figure 3.5-3 indicates, the use of cross feeding leads to lower engine package weights than when cross feeding is not used. This is due to the lower overall vehicle mass and the subsequently reduced engine thrust requirements for the cross fed vehicles.

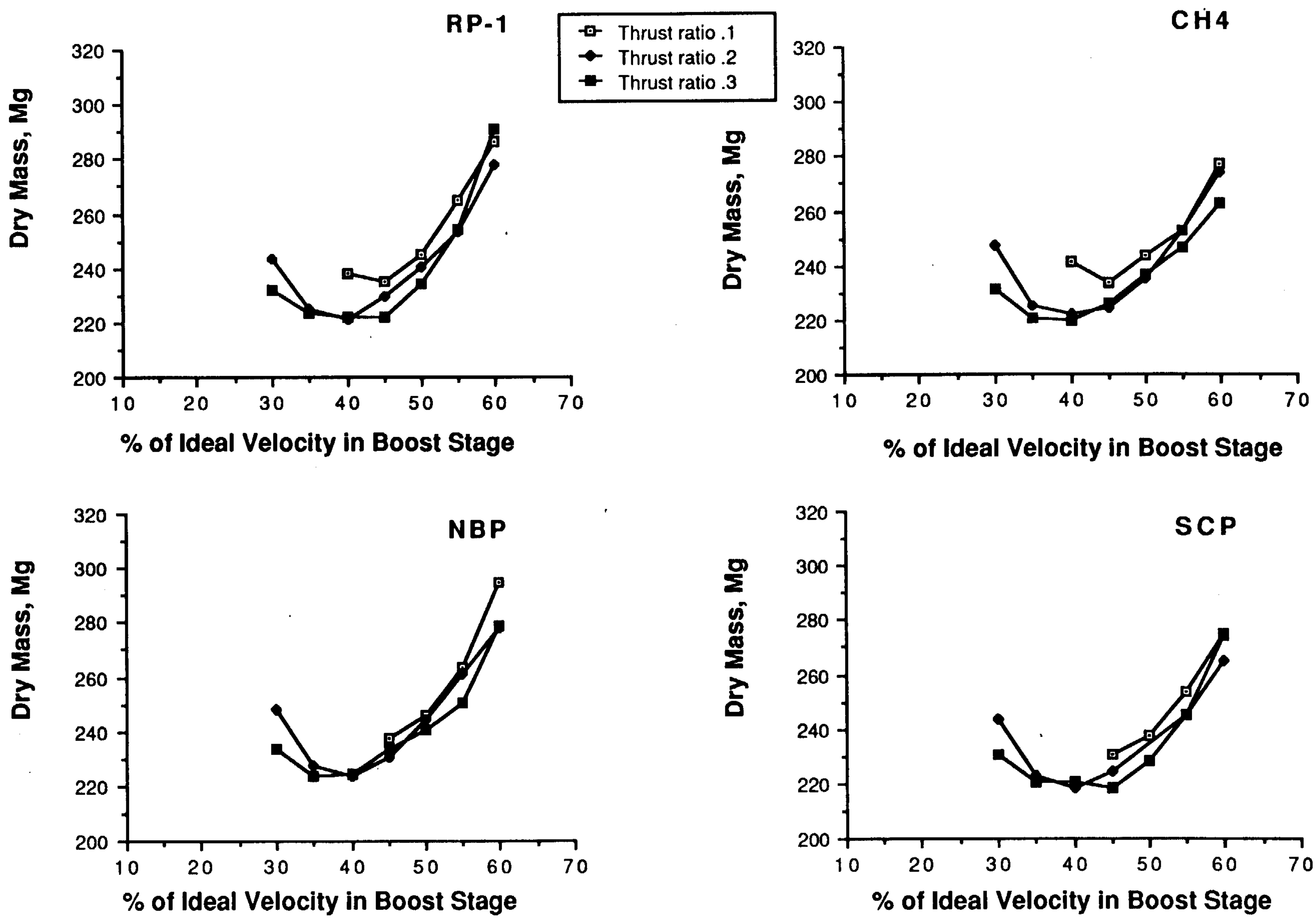


Figure 3.5-1 Total Vehicle Dry Mass for Cross Fed UFRCVs versus Boost Phase Duration for Different Thrust Ratios.

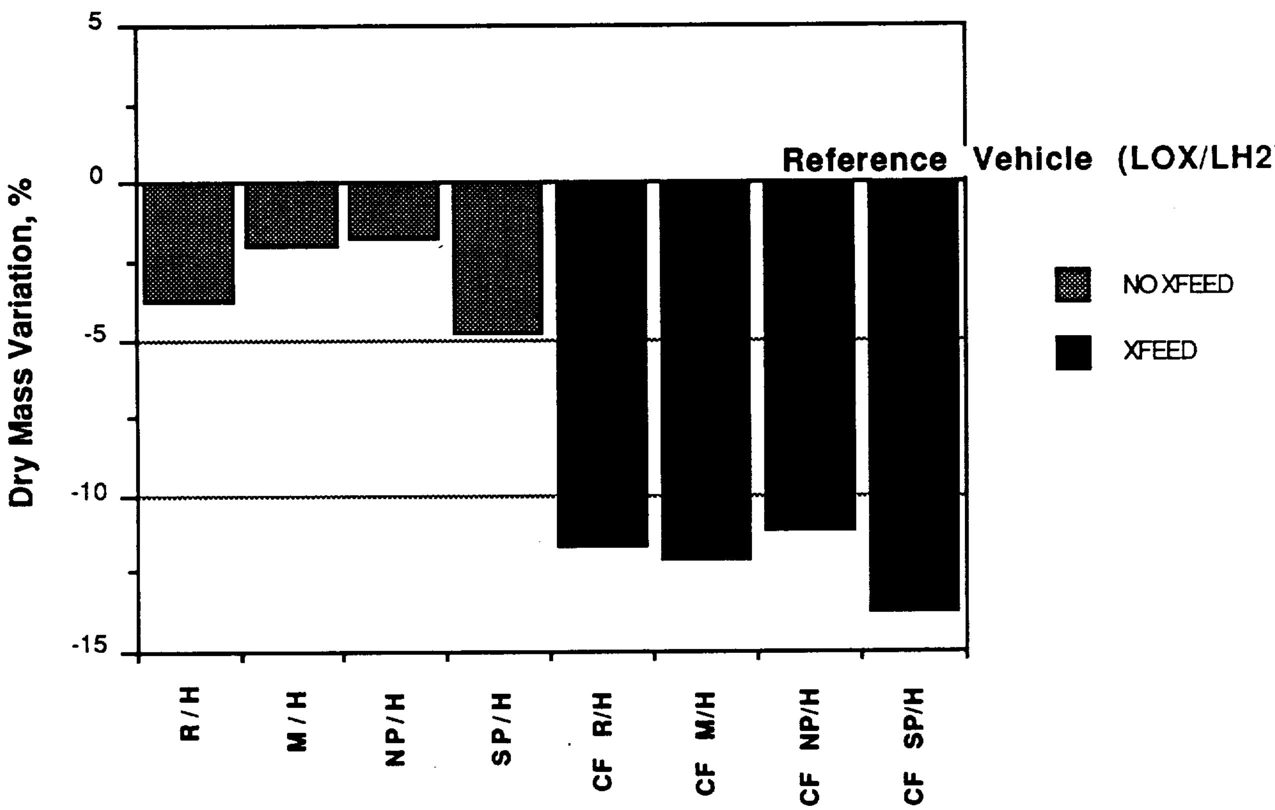


Figure 3.5-2 Comparison of Optimum Cross Feed UFRCVs with Configurations w/o Cross Feed - Total Vehicle Dry Mass

As was expected, the increase in feed system weights when cross feeding propellants is illustrated in Figure 3.5-4. This figure shows that all four optimum configurations with cross feed had values of propulsion subsystem mass, which includes feed and pressurization systems, in excess of both their counterparts without cross feeding and the reference vehicle.

As a final note, Figure 3.5-5 demonstrates that although the use of cross feeding propellants improved weight reductions for the vehicles using hydrogen cooled engines, the final reductions from the reference vehicle were only slightly better than those provided by vehicles using fuel cooling, with the significant exceptions for RP-1 and NBP.

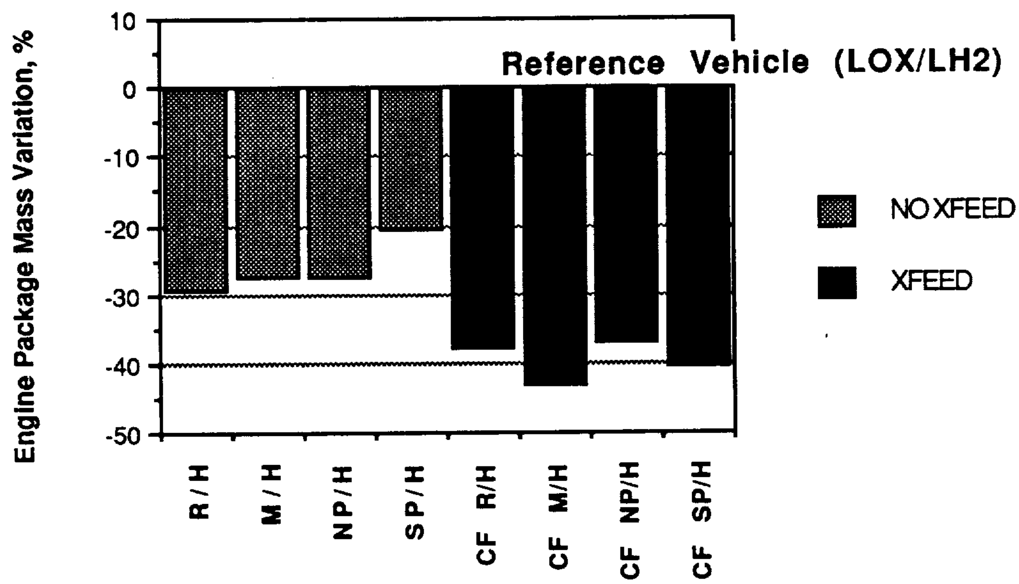


Figure 3.5-3 *Comparison of Optimal Cross Fed UFRCVs to Configurations w/o Cross Feed - Booster Engine Package Mass*

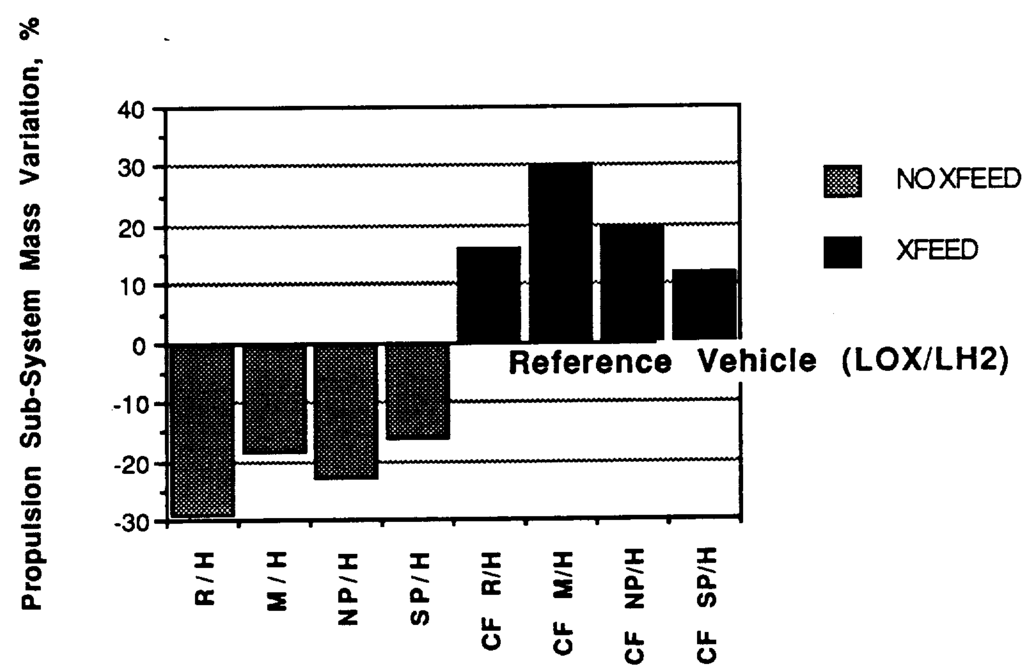


Figure 3.5-4 *Comparison of Optimal Cross Fed UFRCVs to Configurations w/o Cross Feed - Propulsion SubSystem Dry Mass.*

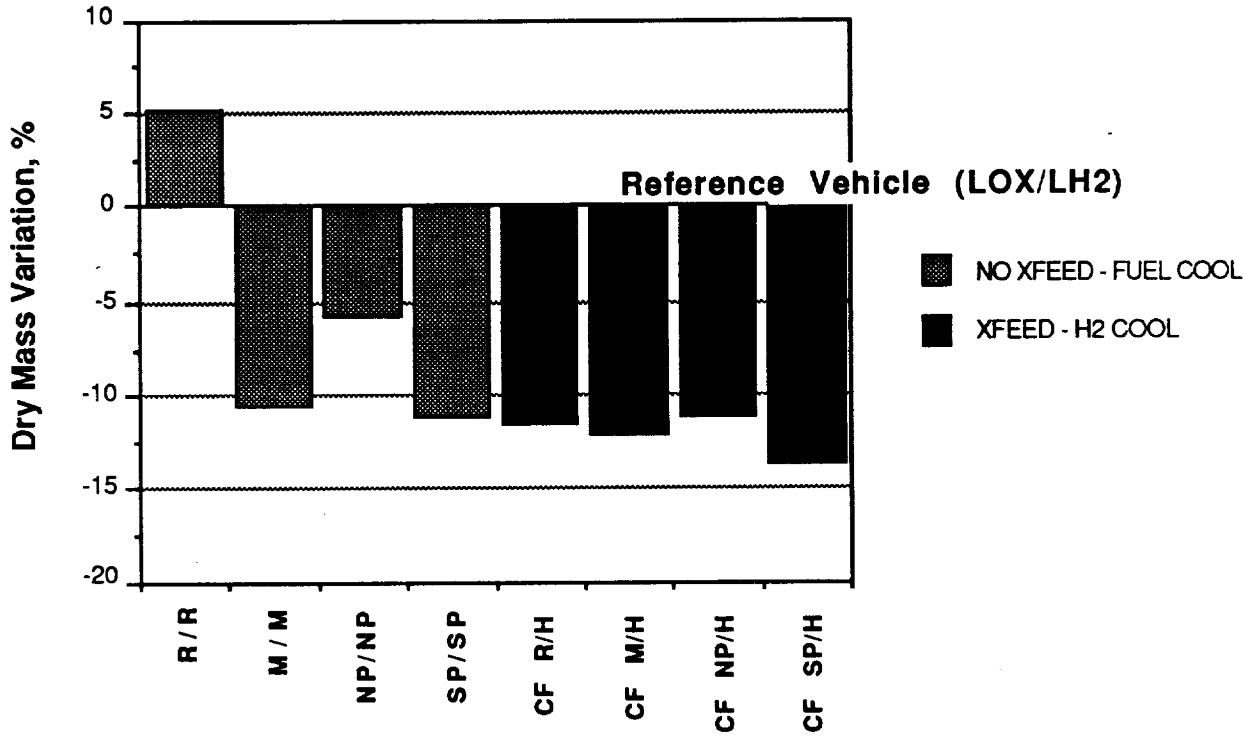


Figure 3.5-5 Comparison of Optimum Cross Feed UFRCVs with Configurations w/o Cross Feed and Fuel Cooled Engines - Total Vehicle Dry Mass

3.6 Task 1.4 - High and Variable Mixture Ratio for All Hydrogen Vehicles

3.6.1 Objective

The objectives of this subtask were: a) determine the impact on the two vehicle types of using high mixture ratio LOX/LH2 engines during the boost phase of flight and b) determine the impact on the two vehicle types when a variable mixture ratio LOX/LH2 engine is used during the boost phase. A variable mixture ratio engine (VMRE) can have its engine mixture ratio, and associated engine performance, step-changed during flight. In this analysis, the VMRE is assumed to have only one step change, such as from 10 to 6.

3.6.2 Summary of Task Activity

3.6.2.1 HMRE

The high mixture ratio engine analysis was conducted first. The reference vehicle input files and LOX/LH2 engine data, see Table 3.4-2, for different mixture ratios were used in the analysis. The sizing proceeded by assuming that the HMRE would be used as the boost phase engine in the SSTO, operating in parallel with the sustainer phase engines, and as the booster engine for the UFRCV. Optimization on boost duration and thrust fraction, for the SSTO, and thrust ratio, for the UFRCV, was conducted. Optimum configurations were compared to the referenced vehicles and the optimum vehicles found in Subtask 1.2.

3.6.2.2 VMRE

The VMRE impact analysis was more complex due to additional optimization parameters involved with the VMRE. The two new parameters are: a) the initial and final mixture ratios for the VMRE and b) when the step change in mixture ratio occurs during the boost phase. In order to limit the complexity, some assumptions were made about these and other sizing optimization parameters. Sizing proceeded in the usual manner. Again, optimum configurations for both vehicle types were selected from the results and compared to: the reference vehicles, the optimum HMRE vehicles, and the optimum vehicles that used the hydrocarbon engine options.

3.6.3 Discussion of Analysis Procedure

3.6.3.1 Ground Rules and Assumptions Used

HMRE

The sizing ground rules previously established for Task 1.0 were used in this analysis. However, for the SSTO analysis it was assumed that a high mixture ratio engine (HMRE) is used during the boost phase and a LOX/LH2 engine with mixture ratio of 6 operates during the sustainer phase, with an expansion ratio of 150. This assumption is unlike the reference vehicle analysis where the same engine was assumed to operate during both the boost and sustainer phases of flight for the SSTO. The HMRE and sustainer phases engines are assumed to operate in parallel at lift-off, although the mixture ratio 6 engine has a lower expansion ratio during the boost phase. The HMRE has an exit pressure of 41.4 KPa as established by the sizing ground rules. For the UFRCV, the HMRE is the engine used in the booster. Thus the analysis for the booster is the same as that conducted for the reference vehicle where engines of different mixture ratios are used in the booster but the orbiter engine remains fixed as a LOX/LH2 engine with a mixture ratio of 6.

VMRE

The added optimization parameters required for the VMRE analysis made it necessary to simplify the assumptions used for the other sizing parameters of thrust fraction and thrust ratio. Therefore, the thrust fraction for the SSTO and the thrust ratio of the UFRCV were assumed to be the same as for the optimum cases found for the HMRE analysis. In addition, rather than examine multiple combinations of initial and final mixture ratios that were possible, the number of cases to be examined were restricted to four for the SSTO and five for the UFRCV. It was assumed that the final mixture ratio for the VMRE was to always be that selected for the reference vehicle, ie. mixture ratio 8 for the SSTO and mixture ratio 7 for the UFRCV. Thus the four cases, defined as initial to final mixture ratio, for the SSTO were: 10 to 8, 12 to 8, 14 to 8 and 18 to 8. The five cases for the UFRCV were similar, with the final mixture ratio 7 rather than 8, and with the addition of the case of 8 to 7. It was decided to optimize on boost phase duration and the fraction of the boost phase that the VMRE was in effect, thus establishing when the step change occurred. It was further assumed that the VMRE was employed in a similar manner as the HMRE for the SSTO, ie. used in parallel with the sustainer engine of mixture ratio 6 as described above.

3.6.3.2 Input Information

The input required for this task included the reference vehicle input files and the engine data. The engine data for the HMRE analysis was obtained from the previously supplied LOX/LH2 engine data, see Subtask 1.1 and Table 3.4-2. A summary of the data used for this analysis is shown in Table 3.6-1. However, there was a lack of data on VMREs. In discussions with the customer, it was decided that a combination of the already supplied LOX/LH2 engine data would be used as VMRE data . It was assumed that the engine performance of the VMRE at high mixture ratio matched the performance of a typical LOX/LH2 engine with that same mixture ratio. When the mixture ratio shifted, it was determined that the engine performance, in terms of specific impulse, of a LOX/LH2 engine with the lower mixture ratio was valid. This procedure implies a constant engine specific impulse efficiency when changing mixture ratios. This assumption leads to specific impulse values larger than an actual engine design for variable mixture ratio. Thrust provided by the VMRE at the low mixture ratio would be calculated based upon assuming that the mass flow rate of fuel was the same as for the high mixture ratio point, this assumes that the mixture ratio is altered by lowering oxidizer flow to the engine. The engine weight would be determined by using the parametric data supplied. Whatever mixture ratio operating point had the highest engine weight was used. The resulting engine data used in this task is shown in Table 3.6-2.

3.6.3.3 Procedures

HMRE

The analysis conducted was slightly different for the two vehicle types due to the ground rules established for this task. The sizing for the SSTO using HMREs proceeded in the same manner as the parallel burn mode analysis in Subtask 1.2. Optimization was conducted on both the thrust fraction and boost duration parameters. The former defined as the fraction of total thrust supplied by the HMRE and the latter as the fraction of total vehicle propellant burned by the HMRE. The thrust fraction parameter was varied from 35% to 75% while the fraction of total vehicle propellant was varied from .2 to .4. Runs were conducted for HMREs using mixture ratios 10 and 12. After conducting the mixture ratio 12 sizing the total dry weight trend indicated that the use of higher mixture ratios, ie. 14 and 18, would only generate heavier vehicles. Total dry weights for each vehicle generated were plotted against the optimization parameters in order to establish the optimum configurations.

The UFRCV analysis proceeded in the same manner as for the reference vehicle analysis although the range of mixture ratios was extended. Here the analysis used mixture ratios of 10 and 12 as for the SSTO. Higher mixture ratios were not justified as the trend of larger total vehicle dry weights as mixture ratio increased was clear. Optimization was done on booster duration (staging velocity) by varying the ideal velocity fraction from .35 to .7 in .05 increments. A thrust ratio of .2 proved to be optimum for the reference vehicle analysis and that same value was used for this analysis. Resulting weights were plotted

Table 3.6-1 High Mixture Ratio Engines

SSTO

MR	ISPvac (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
10	396.57	41.6	20.7	2224	2726	2.362
12	369.76	41.6	20.7	2224	2768	2.369
14	347.85	41.6	20.7	2224	2817	2.380
16	329.43	41.6	20.7	2224	2869	2.407
18	313.65	41.6	20.7	2224	2913	2.439

UFRCV

MR	ISPvac (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
10	396.57	41.6	20.7	3336	4698	3.544
12	369.76	41.6	20.7	3336	4784	3.553
14	347.85	41.6	20.7	3336	4894	3.571
16	329.43	41.6	20.7	3336	5009	3.610
18	313.65	41.6	20.7	3336	5123	3.659

Table 3.6-2 Variable Mixture Ratio Engine Data

SSTO

MR	Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
10.0 / 8.0	396.57 / 428.36	41.6	20.7 / 19.2	2224.1 / 1965.6	2727.3	2.362
12.0 / 8.0	369.76 / 427.81	41.6	20.7 / 16.3	2224.1 / 1781.5	2769.2	2.369
14.0 / 8.0	347.85 / 427.57	41.6	20.7 / 14.9	2224.1 / 1640.3	2818.4	2.380
18.0 / 8.0	313.65 / 427.15	41.6	20.7 / 12.8	2224.1 / 1434.8	2915.8	2.439

UFRCV

MR	Isp (sec)	AR	Pc (MPa)	Tvac (KN)	Mass (Kg)	Aexit (m2)
8.0 /7.0	429.07 / 438.94	41.6	20.7 / 16.1	2224.1 / 2022.5	2706.8	2.350
10.0 / 7.0	396.57 / 438.99	41.6	20.7 / 16.5	2224.1 / 1790.6	2727.3	2.362
12.0 / 7.0	369.76 / 438.82	41.6	20.7 / 15.0	2224.1 / 1624.3	2769.2	2.369
14.0 / 7.0	347.85 / 438.66	41.6	20.7 / 13.7	2224.1 / 1495.9	2818.4	2.380
18.0 / 7.0	313.65 / 438.39	41.6	20.7 / 11.8	2224.1 / 1308.9	2915.8	2.439

against the optimization parameters to establish the optimum configurations for each mixture ratio.

VMRE

As for the HMRE analysis, the VMRE investigation for the SSTO was conducted in a similar manner as the parallel burn study in Subtask 1.2. However, the optimum thrust fraction found in the HMRE analysis was used rather than optimize on thrust fraction. Boost duration optimization, by varying the fraction of total propellant that is expended during boost phase from .2 to .4, was conducted. In addition the fraction of the boost phase that the VMRE operated at the high mixture ratio was also varied. This was done by defining a new parameter for the SSTO program. This parameter was the fraction of the total boost phase propellant expended by the VMRE during high mixture ratio operation. This fraction was varied from .2 to .5 in .1 increments. This optimization was applied for each variation in boost duration. Thus 12 optimization points were generate for each of the four VMRE cases for a total of 48 runs. As for the HMRE, the resulting vehicles were plotted against the optimization parameters to identify the optimum configurations for each VMRE case and then compared to the previously determined optimum HMRE cases, the reference vehicle and to the optimum configurations using the hydrocarbon engine options.

The UFRCV analysis was conducted by using the assumption that the thrust fraction was fixed to the optimum value determined during the HMRE analysis. To optimize on booster duration, the booster fraction of total ideal velocity from .4 to .6 was varied in .05 increments. This range, which is more narrow than previous analyses, was selected based upon the results for the HMRE analysis. As for the SSTO, a new optimization parameter was defined to allow optimization for the VMRE's high mixture ratio duration. This parameter was defined as the fraction of the booster ideal velocity provided by the VMRE operating in high mixture ratio mode, the remainder assumed to be provided when the VMRE was in the low mixture ratio mode. It was decided to vary this fraction from .1 to .5 in .1 increments and alter the range and increment value based upon intermediate results. Assuming a full range to be required for each booster duration point implied a total of 25 runs for each VMRE case, or a total of 125 runs. This large number was reduced during the study by reducing both the boost duration range and high mixture operating mode range as intermediate results indicated. Resulting total vehicle dry weights were plotted against the optimization parameters to identify the optimum configurations. As for the SSTO, these optimum configurations were compared to other UFRCVs.

3.6.4 Discussion of Analysis Results and Conclusions

3.6.4.1 HMRE

Figures 3.6-1 and 2 illustrate the results for the SSTO configuration. The Tfrac variable is the thrust fraction. In these figures the boost phase duration is indicated by the percentage of total vehicle propellant burned by the HMRE

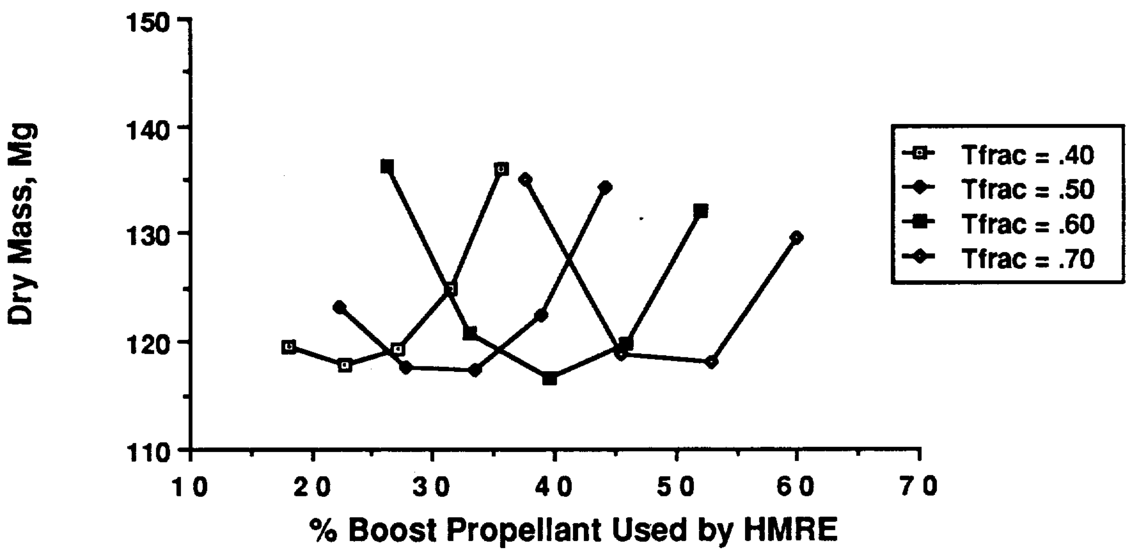


Figure 3.6-1 *Total Vehicle Dry Mass for SSTO versus Boost Phase Duration for HMRE with Mixture Ratio 10 for Different Thrust Fractions*

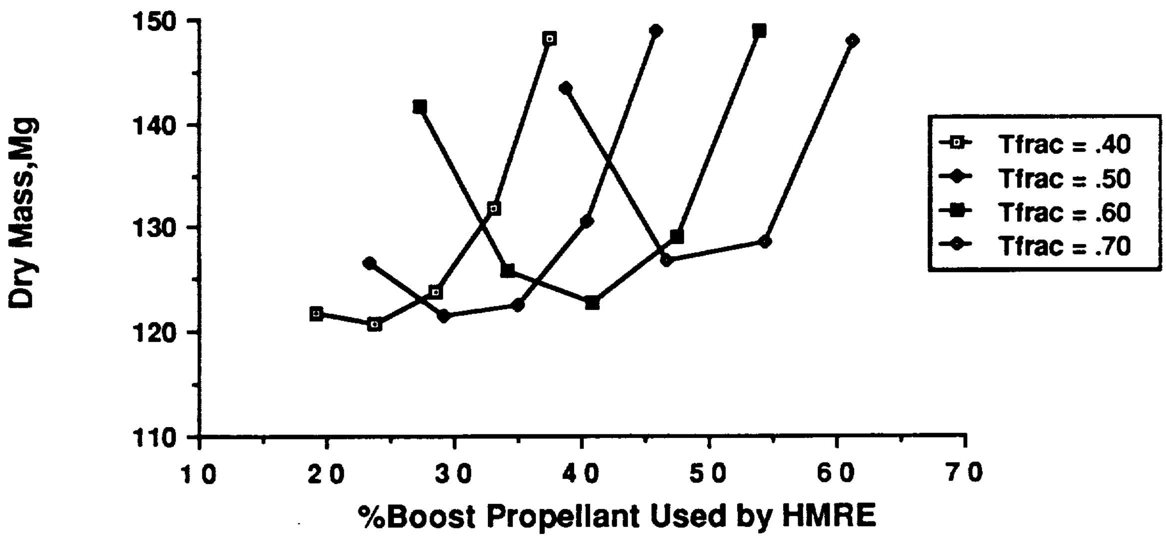


Figure 3.6-2 *Total Vehicle Dry Mass for SSTO versus Boost Phase Duration for HMRE with Mixture Ratio 12 for Different Thrust Fractions*

engines. Figure 3.6-1 shows that for an engine operating at a constant mixture ratio of 10 a minimum dry weight point occurs for a Tfrac value of .6 when the percentage of total propellant expended by the HMRE engines is 40%. However, this minimum point, a dry mass value of 116 Mg, is larger than the reference vehicle. For an HMRE operating at a mixture ratio of 12, a minimum point is not indicated on the graph. What is of note is that as the Tfrac value decreases the percentage of propellant expended by the HMRE engines and the vehicle dry mass decreases. This trend is interpreted to mean that the vehicle wants to use as little of the HMRE, for thrust as to expend propellant, as possible. As the thrust provided and propellant expended by the HMRE decreases the total vehicle dry mass approaches the reference vehicle value. In essence, the minimum dry mass value would be found for a Tfrac value and propellant expended value of 0.

The trend demonstrated for an HMRE operating at a mixture ratio of 12 was more severe for the initial sizing of an SSTO using an HMRE operating at a mixture ratio of 14 and thus did not allow optimization for this case. In all the cases, the total vehicle dry mass exceeded that of the reference vehicle by 20 percent or more. Figure 3.6-3 shows the comparison between the minimum dry mass points for an HMRE operating at a mixture ratio of 10 and 12 to the reference SSTO vehicle. The use of an HMRE of any kind in an SSTO of the baseline design only increases total vehicle dry mass when compared to a LOX/LH2 engine operating at near optimum mixture ratio.

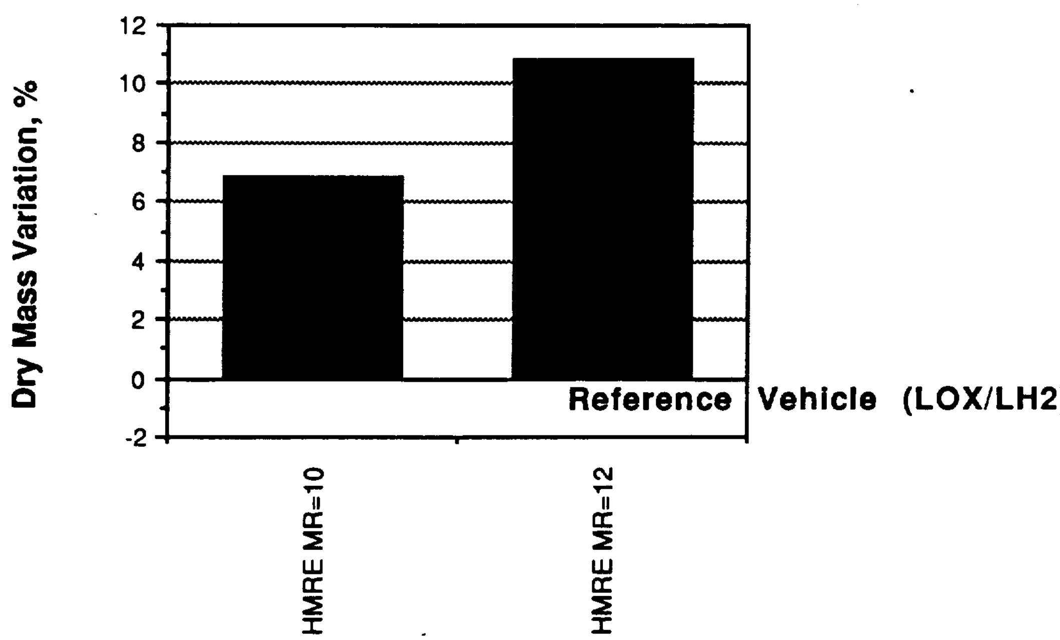


Figure 3.6-3 Comparison of Optimum SSTOs Using High Mixture Ratio Engines - Total Vehicle Dry Mass

Figure 3.6-4 illustrates the total vehicle dry mass values for the UFRCV for engines operating at three different mixture ratios over a range of booster duration, as before defined by the percentage of ideal velocity in the boost stage. The HMRE analysis results for mixture ratio 10 and 12 are shown as compared to the reference vehicle results from subtask 1.2. Note that the HMRE options always generate greater vehicle dry mass than for an engine operating a near optimum mixture ratio. Also notable is the minimum point for each curve is at a different value for percentage of ideal velocity in the boost stage. In fact, as the mixture ratio of the HMRE goes up the minimum point shifts to the left on the graph. As for the SSTO, this trend is interpreted to indicate that as the mixture ratio value increases the vehicle seeks to use the booster less and less in order to offset the specific impulse penalty and to achieve a minimum dry mass. Comparison of the minimum dry mass vehicles to the reference vehicle, for the mixture ratio values 10 and 12, is made in Figure 3.6-5. It is apparent that the use of an HMRE as a booster engine for the UFRCV provides no mass reduction as compared to the reference vehicle.

3.6.4.2 VMRE

Figure 3.6-6 shows the results of the VMRE analysis for the SSTO. Total vehicle dry mass is shown for the different VMRE cases versus boost phase duration and duration of VMRE operation at high mixture ratio values. The boost phase duration is shown as the percentage of total vehicle mass used as boost phase propellant. The VMRE has two modes of operation. Mode one is when the engine operates at high mixture ratio. The duration of mode one is controlled by varying the fraction of boost phase propellant used when the VMRE is in mode one. PB1 is the variable used and it represents the fraction of boost phase during which the VMRE is in mode one, the high mixture ratio value. Various values of PB1 are shown in the legend for the figure. All the VMRE cases optimized total vehicle dry mass with a PB1 value of .55. The line curve representing the PB1 equals .55 for each VMRE case was plotted on the same graph as shown in Figure 3.6-7. It is clear that as the high mixture ratio value for the VMRE increased so did the total vehicle dry mass. From Figure 3.6-7, minimum vehicle dry mass points were identified for each VMRE case. These were considered the optimum vehicles for the four VMRE options. These optimum vehicles are compared to the reference vehicle in Figures 3.6-8 and 9 on a total vehicle dry mass and propulsion system mass basis respectively. All of the VMRE options generated vehicles with total vehicle dry mass values in excess of that for the reference case. The smallest increase is 5 percent for the VMRE option 10 to 8. This option also had slightly lower propulsion system mass than the reference vehicle.

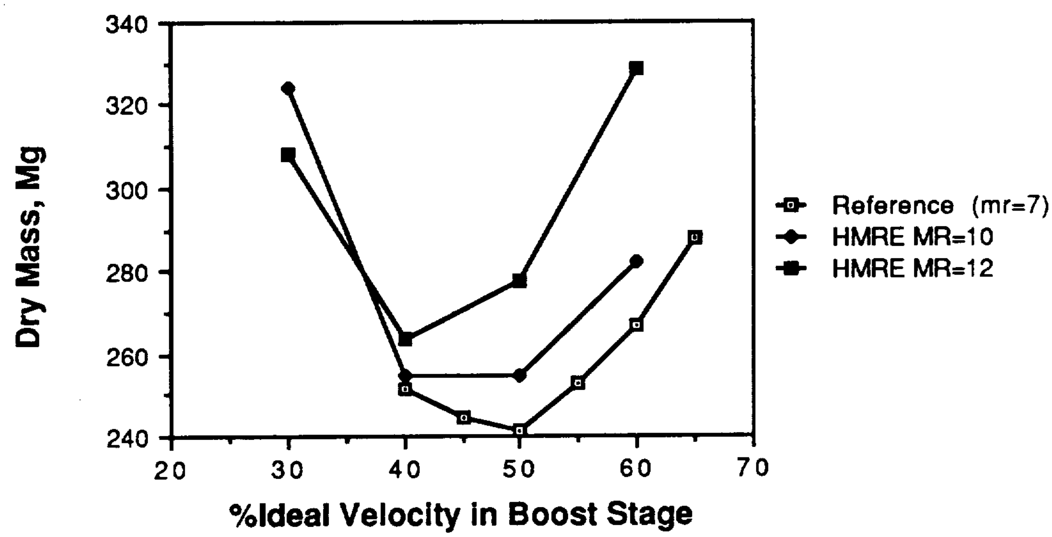


Figure 3.6-4 Total Vehicle Dry Mass Versus Boost Duration for UFRCVs Using High Mixture Ratio Engines.

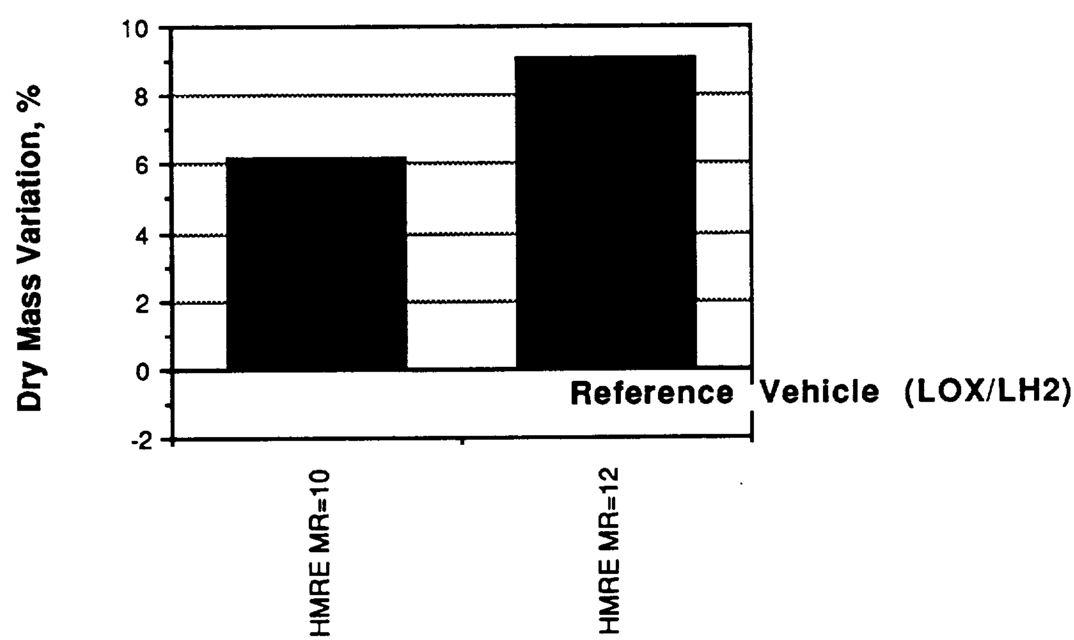


Figure 3.6-5 Comparison of Optimum UFRCVs Using High Mixture Ratio Engines-Total Vehicle Dry Mass

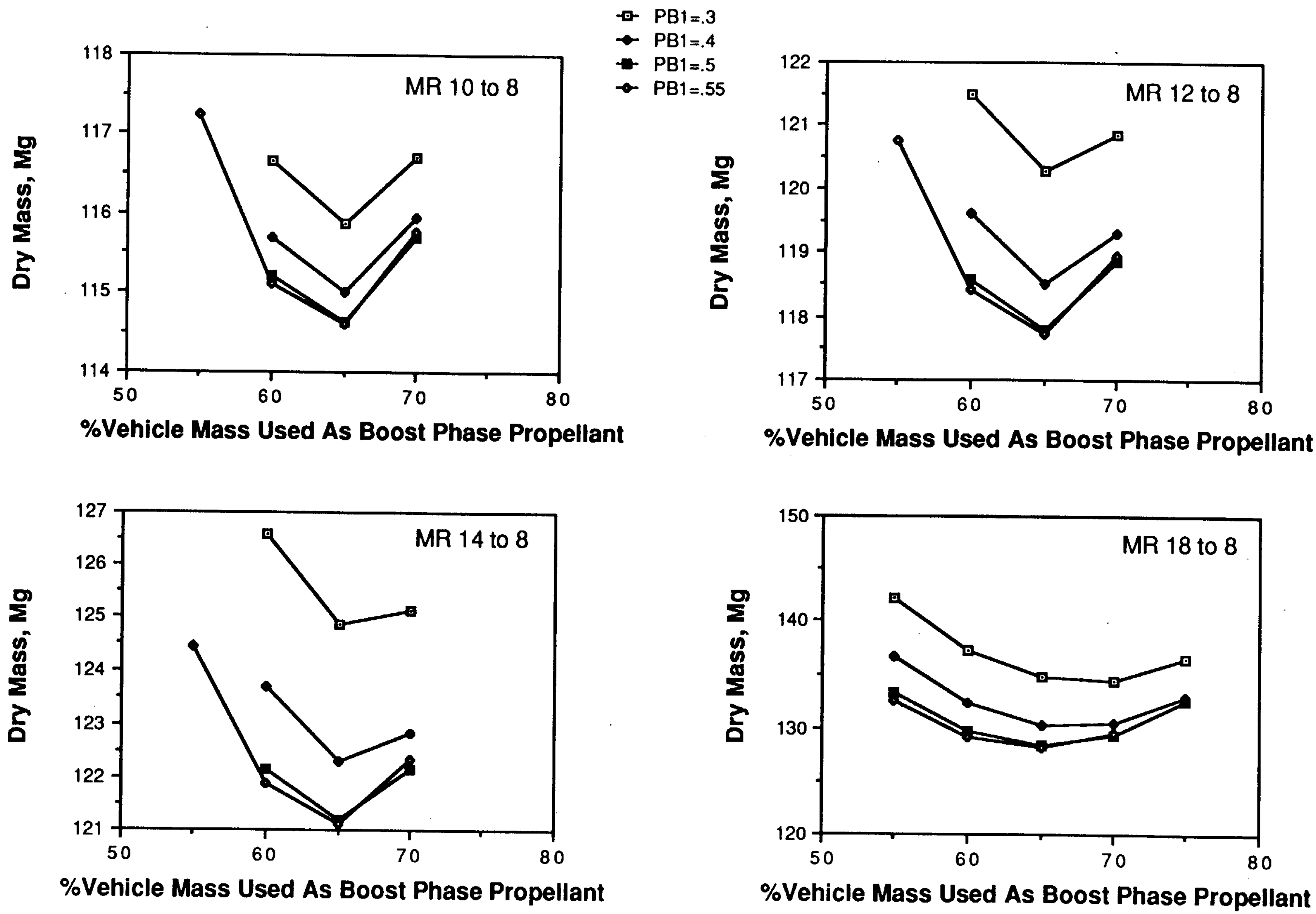


Figure 3.6-6 *Total Vehicle Dry Mass for SSTO versus Boost Phase Duration for VMREs with Different Mixture Ratios and for Different Mode One Durations*

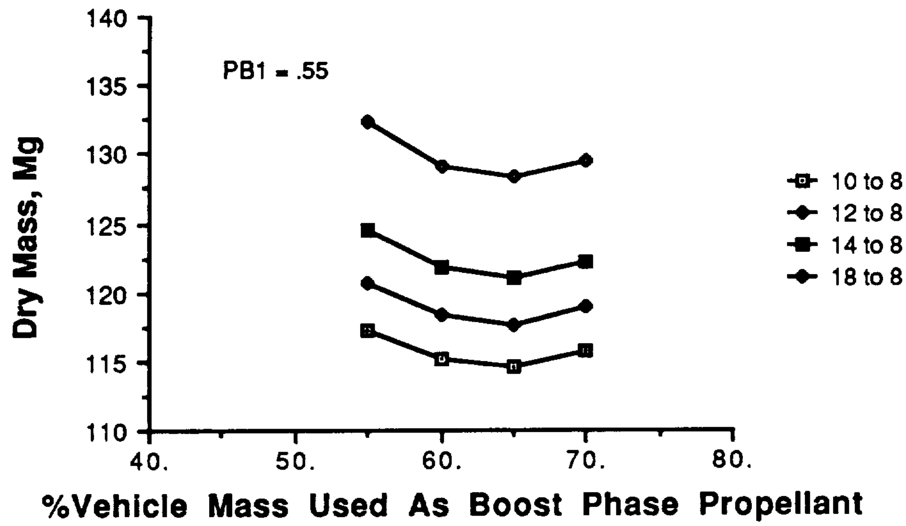


Figure 3.6-7 Total Vehicle Dry Mass Versus Boost Duration for SSTOs Using Variable Mixture Ratio Engines.

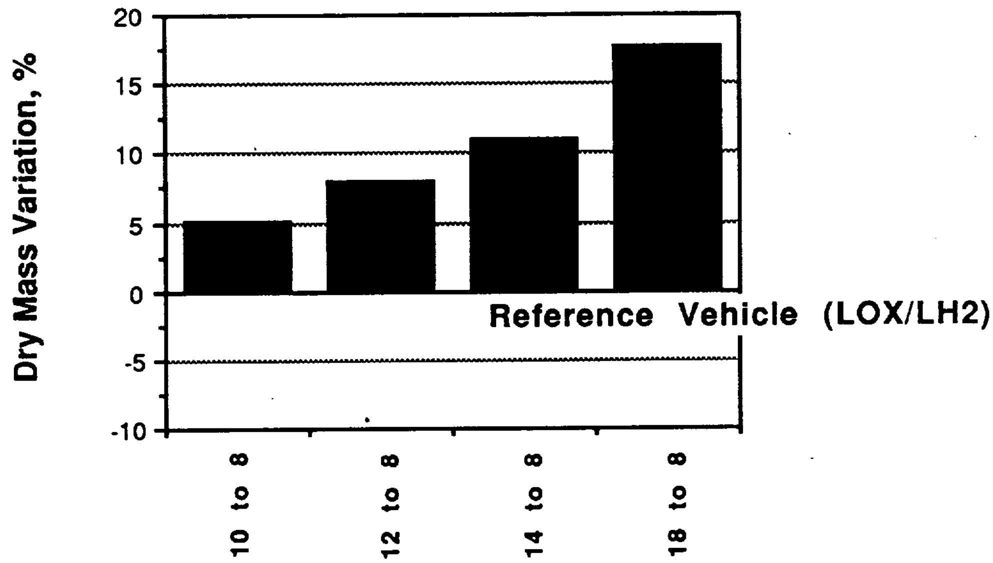


Figure 3.6-8 *Comparison of Optimal Variable Mixture Ratio SSTO Configurations - Total Vehicle Dry Mass.*

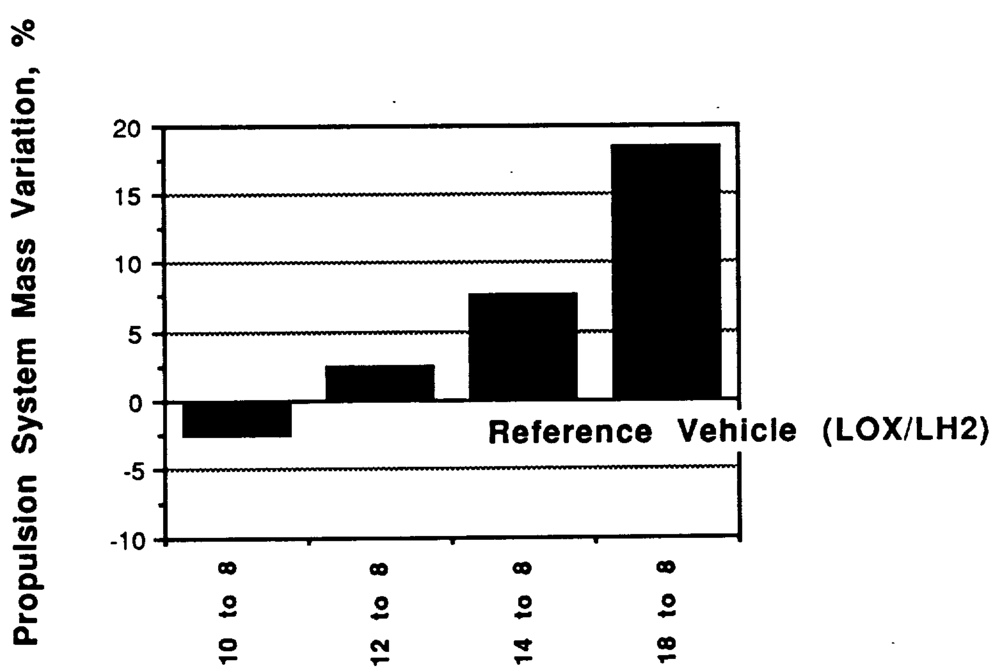


Figure 3.6-9 *Comparison of Optimal Variable Mixture Ratio SSTO Configurations - Propulsion System Dry Mass.*

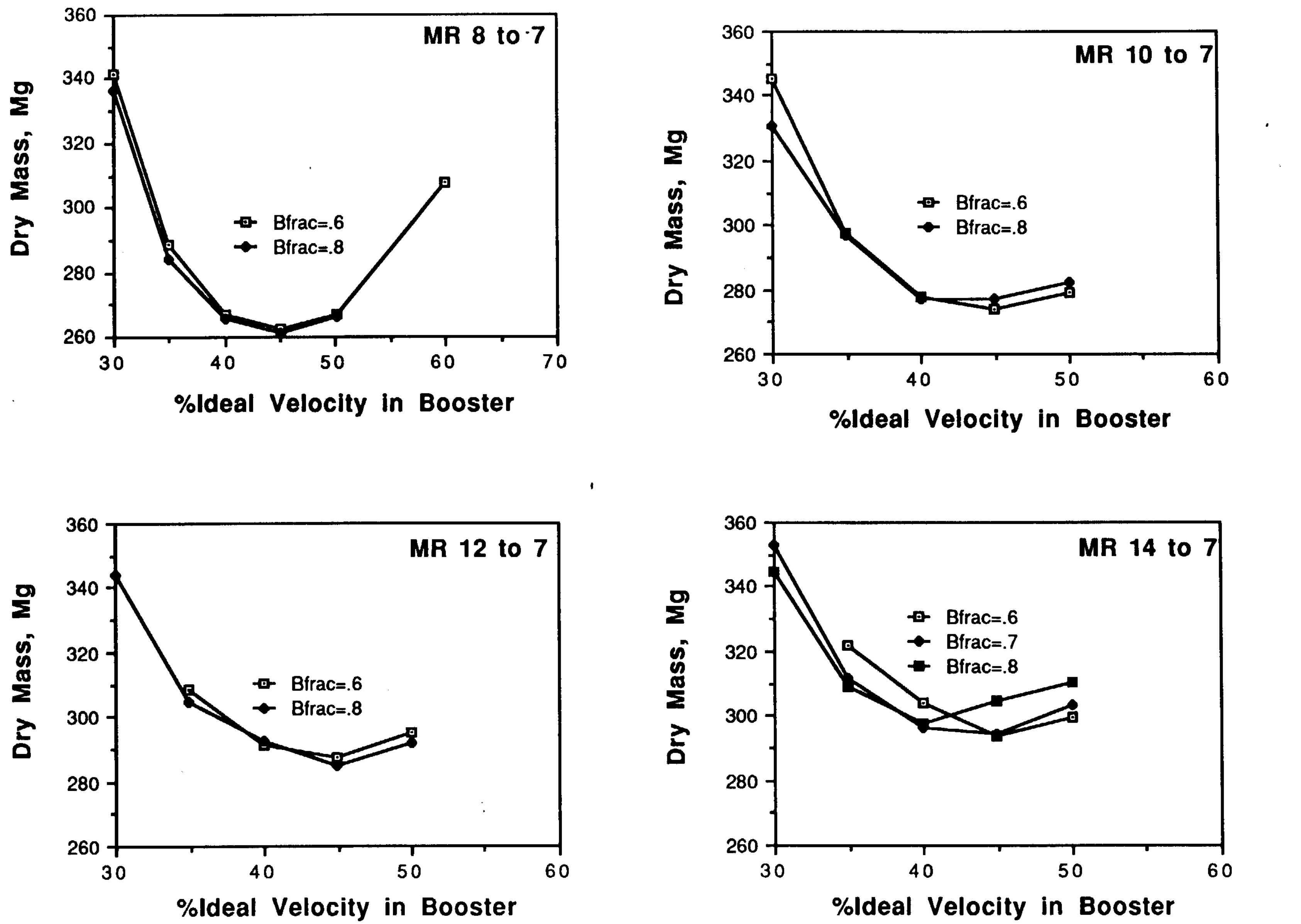


Figure 3.6-10 Total Vehicle Dry Mass for UFRCV versus Boost Phase Duration for VMREs with Different Mixture Ratios and for Different Mode One Durations

Figure 3.6-10 shows the results of the VMRE analysis for the UFRCV. This figure shows total vehicle dry mass versus boost phase duration for the four cases examined for different durations of mode one of the VMRE. In this case, mode one duration is controlled by the variable Bfrac. This variable represents the fraction of boost stage ideal velocity provided by the VMRE during mode one operation. It is analogous to the percentage of ideal velocity in boost stage parameter. Only four of the five original cases were completed due to the extremely large, when compared to the reference UFRCV, vehicle dry mass values that resulted when the case five VMRE option, mixture ratio 18 to 8, was examined. From each graph shown in Figure 3.6-10, the curve that generated the lowest vehicle dry mass was selected, representing a specific Bfrac value. values. The four curves were all plotted on the same graph which is shown in Figure 3.6-11. As for the SSTO, the clear trend is that as the mixture ratio of mode one increases, the total vehicle dry mass values increase. The minimum dry mass points from the four curves were selected to establish the optimum vehicles for use of the VMRE option.

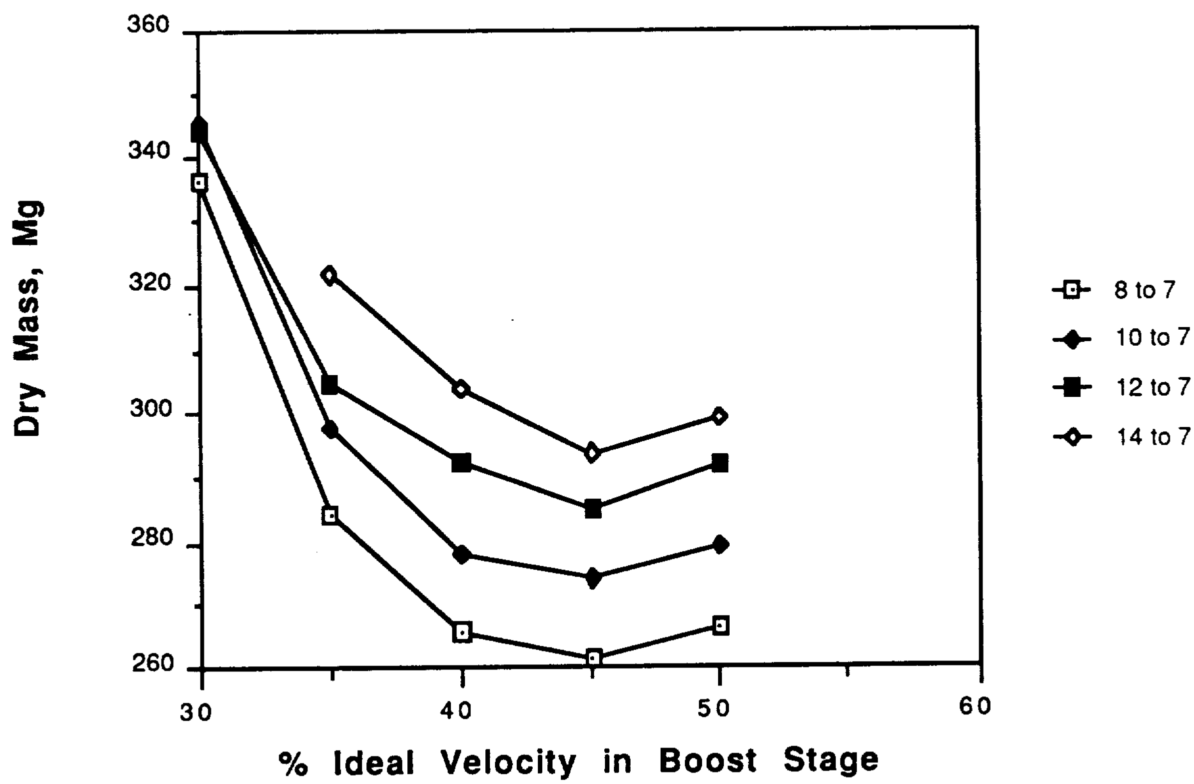


Figure 3.6-11 Total Vehicle Dry Mass Versus Boost Duration for UFRCVs Using Variable Mixture Ratio Engines.

The optimum vehicles selected are compared to the reference UFRCV in Figures 3.6-12 through 14. The first figure compares total vehicle dry mass and it is immediately obvious that all of the VMRE options generate vehicles with greater dry mass than the reference vehicle. The smallest increase is 8 percent. Given the trends indicated in this graph it is estimated that the increase in dry mass for the VMRE case five, which was not completed, would be in excess of 25 percent. The other comparison figures, 13 through 14, further indicate that the VMRE options merely generate bigger and heavier vehicles as compared to the reference vehicle.

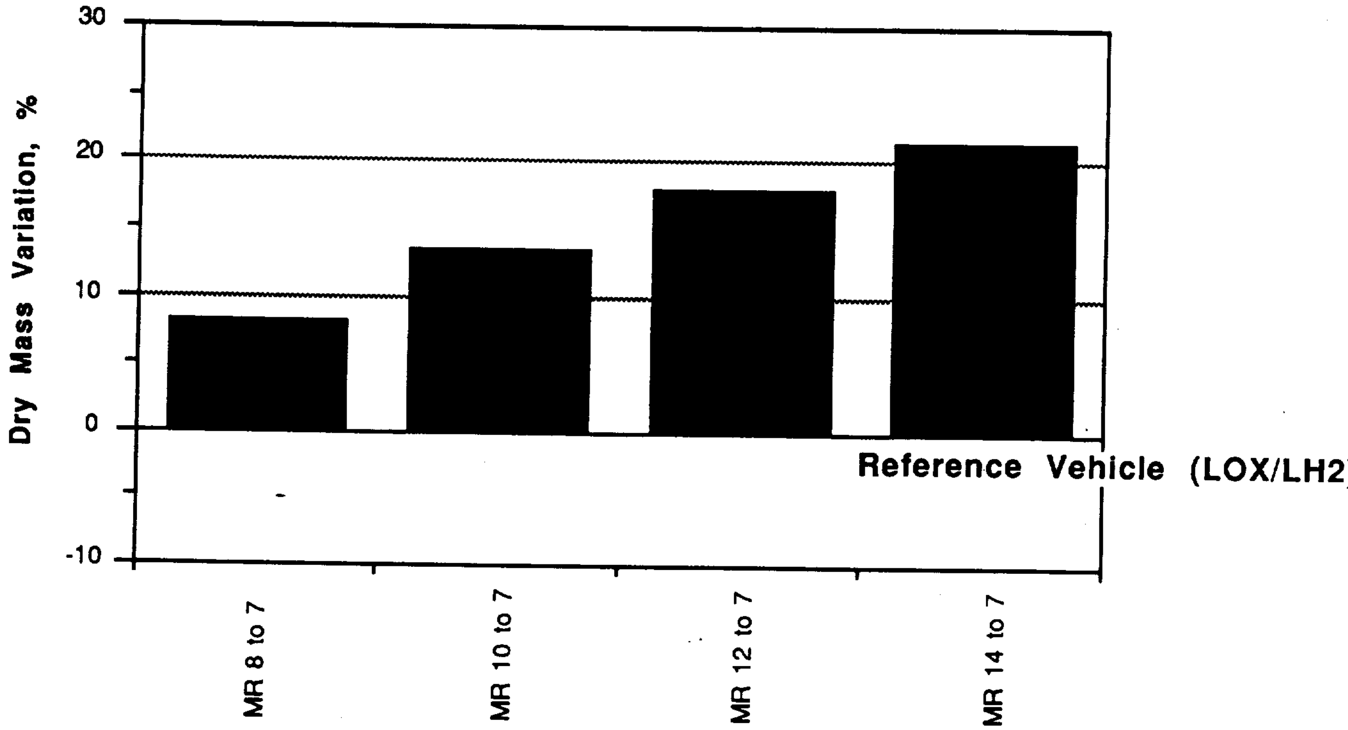


Figure 3.6-12 Comparison of Optimal UFRCVs Using Variable Mixture Ratio Engines-Total Vehicle Dry Mass

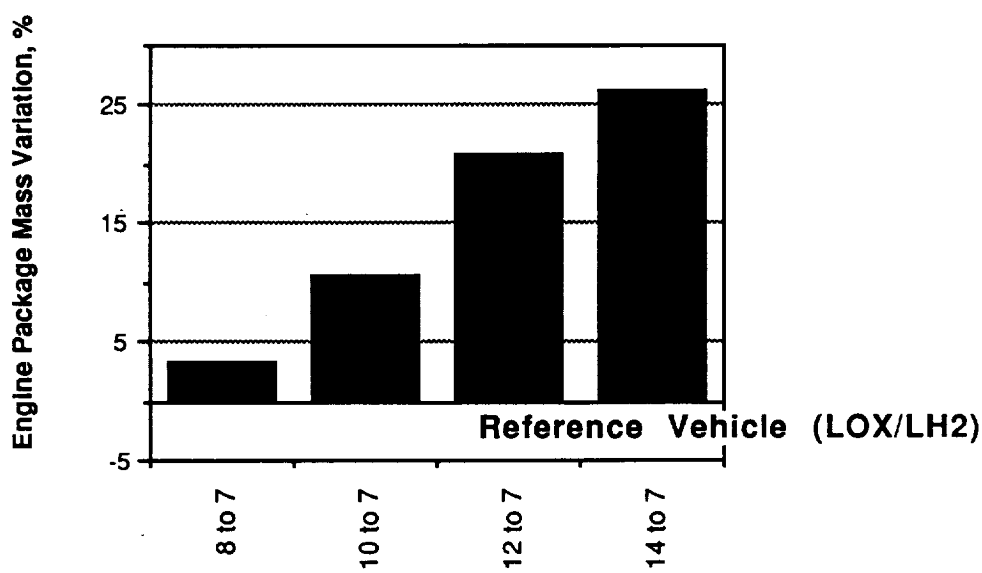


Figure 3.6-13 Comparison of Optimum UFRCV Configurations for VMRE Options- Booster Engine Package Mass.

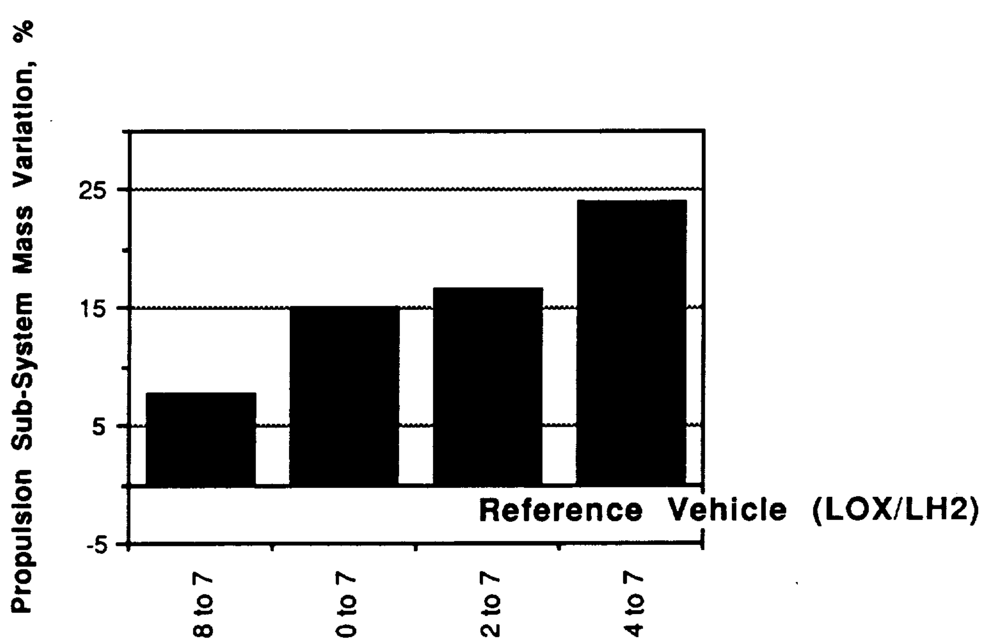


Figure 3.6-14 Comparison of Optimal UFRCV Configurations for VMRE Options - Booster Propulsion Sub-system Mass.

3.7 Task 1.5 - Isp Step Change Impacts (Translating Nozzles)

3.7.1 Objective

The objectives of this task were to determine the impact on the SSTO and UFRCV reference vehicles when a step change in specific impulse occurs in the boost phase engine during the boost phase of flight. This was assumed to be caused by the use of a translating nozzle on the boost phase engine. This analysis was conducted for all the hydrocarbon engine options and for a LOX/LH2 engine for a total of nine engine cases. The latter engine was not used in the SSTO analysis because the reference, all LOX/LH2, SSTO already used a version of a translating nozzle.

3.7.2 Summary of Task Activity

The reference vehicle input files for the sizing models and the LOX/LH2 and LOX/HC engine data were used as input to this task. An assumption was made that only one extendable nozzle case would be examined for both vehicles. Input files were created for the sizing models that investigated boost phase duration and the point in the boost phase when the higher expansion ratio nozzle was extended. Sizing analysis was conducted in the same manner as for the reference vehicles. Optimum configurations identified were compared to the reference vehicles and, when warranted, to the other configurations identified in Subtask 1.2.

3.7.3 Discussion of Analysis Procedure

3.7.3.1 Ground Rules and Assumptions Used

Typical sizing ground rules and assumptions were used in this task as in the other subtasks. It is believed that the impact of a booster translating nozzle engine (BTNE) on the vehicles justified an investigation of boost phase duration, rather than using the optimum points identified for the reference vehicles. In order to simplify the SSTO analysis, it was assumed that: a) the BTNE operated in parallel with the sustainer phase engine, b) the BTNE was not the same engine as the sustainer phase engine, thus adding an extra engine to the stage as compared to the reference vehicle case and c) that the BTNE used the same mixture ratio and initial thrust as was assumed for the reference vehicle. In addition, rather than optimize on thrust fraction, it was assumed that the optimum values of thrust fraction found during the trades analysis was appropriate to use in this investigation. For the UFRCV analysis it was also assumed that the BTNE initial thrust level and mixture ratio were the same as for the reference case. The thrust ratio value, identified as optimum for the reference vehicle, was assumed to be the correct value for this analysis.

For both vehicles it was assumed that the BTNE started the boost phase with an expansion ratio of 41.6 as was used in the reference vehicle analyses. An expansion ratio that generated a 20.7 KPa exit pressure for the higher expansion ratio nozzle of the BTNE was used. This exit pressure value is consistent with engines that operate from liftoff to orbit in parallel burn, two-stage rockets and results in a nozzle expansion ratio value that is sufficiently different enough from 41.6 as to cause some impact on the vehicle. It is also not so drastic a change as to invalidate the method used to determine the engine data, described in more detail in the procedure section. BTNE data was generated for the eight hydrocarbon engine options and two LOX/LH2 engines, one for each mixture ratio value used for the reference vehicles, 7 for the UFRCV and 8 for the SSTO.

In order to determine when during boost the higher expansion ratio nozzle of the BTNE was to be used, a new optimization parameter, that represented the fraction of boost phase duration that the lower expansion ratio nozzle was in effect, was established. This fraction was initially varied from .2 to .8 and adjusted for each vehicle as the intermediate sizing results indicated. For the UFRCV, this fraction was called Bfrac and its value represented the fraction of boost stage ideal velocity provided by the BTNE when operating at its lowest expansion ratio. For the SSTO, this fraction was called PB1Frac and its value represents the fraction of boost phase propellant expended while the BTNE is operating at the low expansion ratio. It is not the fraction of boost phase propellant actually expended by the BTNE since this engine operates in parallel with the sustainer engine.

3.7.3.2 Input Data

Using the BTNE assumptions discussed above, the previously supplied engine data for LOX/LH2 engines and the data on LOX/HC engines from Reference 1 was used to establish engine characteristics for the BTNE for use on the SSTO and the UFRCV. The specific engine data, generated for the reference vehicle analyses, was used to determine the thrust and specific impulse of the BTNE at the lower expansion ratio. Engine parameters had to be calculated in order to determine the engine characteristics at the higher expansion ratio.

Different methods were used to determine the key engine characteristics of engine mass, engine thrust and specific impulse and expansion ratio. Engine area ratio and delivered specific impulse at the higher expansion ratio point, characterized by the desired exit pressure, were determined using the Air Force Rocket Propulsion Laboratory specific impulse program[12](AFRPL/ISP). The use of this program implies the assumption that the engine specific impulse efficiency value is the same for both expansion ratio points. Hydrocarbon engine mass increases, due to an extendable nozzle, were determined using the WTNOZ (extendable nozzle weight) equations on page 310 of Reference 1; these equations were assumed to include extendable nozzle, actuators and any other additional equipment required for the extendable nozzle. Data was lacking to allow direct determination of engine mass for hydrogen engines with extendable nozzles. It was assumed that the thrust to weight for hydrogen engines was proportional to the thrust to weight for hydrocarbon engines and this proportionality was assumed to be constant for either baselined or extendable nozzle engines Using this proportionality constant, the engine mass for LOX/LH2 BTNEs were determined. Engine thrust levels for the higher expansion ratio were determined using the calculated specific impulse values and an assumption of constant mass flow rate for the engine. The engine data generated using the above procedures and used for this task is shown in Table 3.7-1.

The remainder of the input data consisted of the reference vehicle input files and the assumptions on optimization used for this subtask.

3.7.3.3 Procedure

The calculated engine data was used to generate new sizing input files. A series of files were created to investigate the vehicle impact of varying boost duration and fraction of the boost phase during which the BTNE was operating with the lower expansion ratio. Using the typical ranges of boost duration established during the reference vehicle analyses, a total of 160 files each, for the SSTO and UFRCV sizing models, were generated. These files were processed in the typical manner. The resulting vehicle dry masses were plotted against the boost duration and nozzle extension time parameters in order to establish the configuration optimums. The optimum configurations were then compared to the respective reference vehicles.

Table 3.7-1 Translating Nozzle Engine Data

FUEL	COOLANT	% H_2	MR	AR	Pc (MPa)	Isp(sec)	Tvac (KN)	Mass (Kg)	Aexit (m2)
H2	H2	12.5	7.0	41.6/75.9	20.7	439.9/451.2	2224/2281	3238	2.36/4.31
H2	H2	11.1	8.0	41.6/76.8	20.7	429.1/442.1	2224/2292	3191	2.35/4.34
RP-1	HC	.00	2.42	28.7/49.1	10.3	312.0/320.0	3199/3283	3672	4.81/8.22
CH_4	HC	.00	3.05	48.2/82.9	20.7	350.0/357.6	3108/3175	3313	3.83/6.59
C_3H_8- NBP	HC	.00	2.62	45.5/78.2	18.6	330.2/337.4	3148/3217	3425	4.07/6.99
C_3H_8- SC	HC	.00	2.70	49.0/84.3	20.7	332.8/340.0	3111/3178	3086	3.88/6.68
RP-1	H_2	1.01	2.82	48.4/83.3	20.7	335.4/342.9	3090/3159	3038	3.71/6.39
CH_4	H_2	1.09	3.53	48.2/82.9	20.7	359.8/367.8	3087/3156	3190	3.69/6.35
C_3H_8- NBP	H_2	1.04	3.13	49.2/84.7	20.7	344.9/352.7	3095/3165	3138	3.77/6.50
C_3H_8- SC	H_2	1.01	3.13	49.0/83.4	20.7	343.9/351.6	3094/3163	3006	3.76/6.39

3.7.4 Discussion of Analysis Results and conclusions

Figure 3.7-1 illustrates typical results for the SSTO analysis, in this case for an RP-1 fueled engine with fuel cooling. The separation between the curves for the different values of PB1Frac, or the fraction of boost phase propellant expended during BTNE operation at low expansion ratio, is very small and was the case for all eight hydrocarbon engine options. The graphs for the other options are not shown due to this feature in the data. However, curves, or PB1Frac values, were selected for each of the eight options that generated the lowest total vehicle mass values. These eight curves are shown in Figures 3.7-2, and 3, fuel cooled and hydrogen cooled options respectively. Readily apparent is the fact that the hydrogen cooled engines always generated vehicles with lower dry mass than for the corresponding fuel cooled engines with the minor exception for subcooled propane. From these eight curves, the minimum dry mass point for each fuel/coolant option was identified. The optimum configurations are compared to the reference vehicle and to the corresponding optimum configurations, where a translating nozzle was not used, in Figure 3.7-4 on a total vehicle dry mass basis. With the one exception for R/R, the use of translating nozzles on the hydrocarbon engines generated vehicles with greater dry mass than when the translating nozzle was not used. In addition, the use of translating nozzle on the hydrogen cooled engines resulted in vehicles with greater dry mass than the reference vehicle.

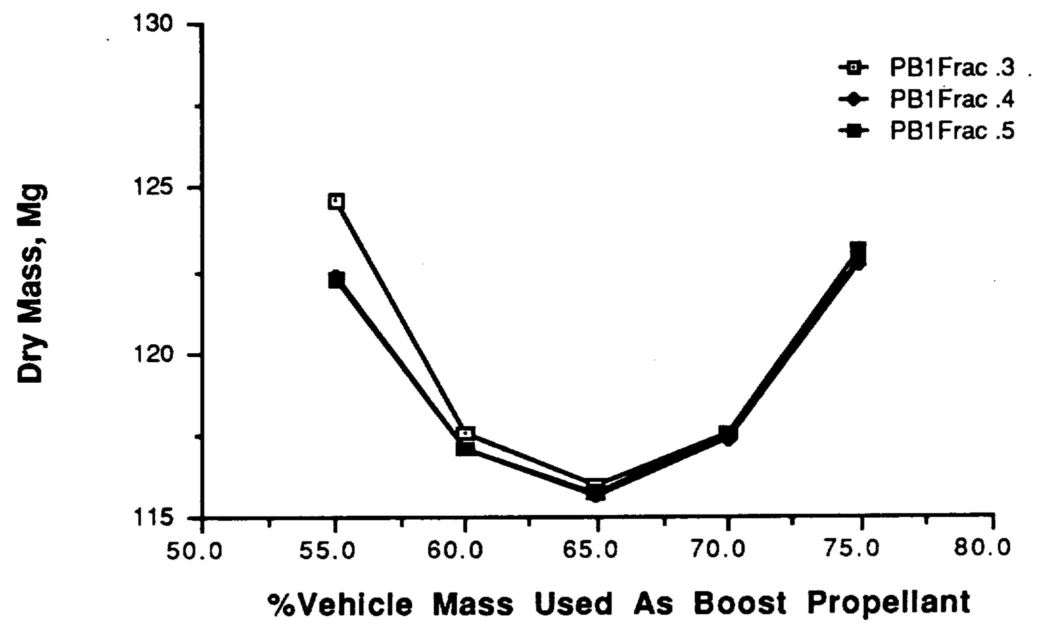


Figure 3.7-1 Typical Results for Dry Mass Versus Boost Duration for an SSTO Using Translating Nozzle-RP-1 Engine With Fuel Cooling

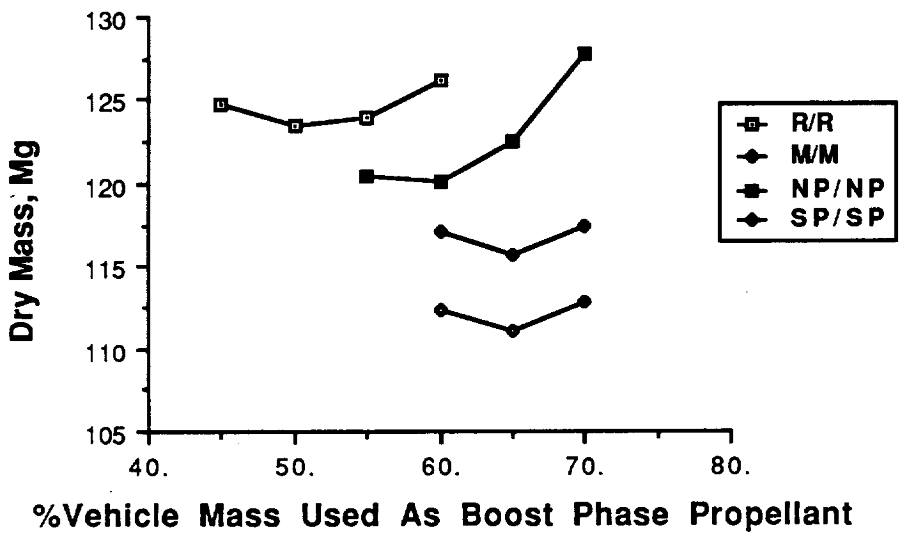


Figure 3.7-2 Dry Mass Versus Boost Duration for SSTO Using Translating Nozzles-Fuel Cooled Engines

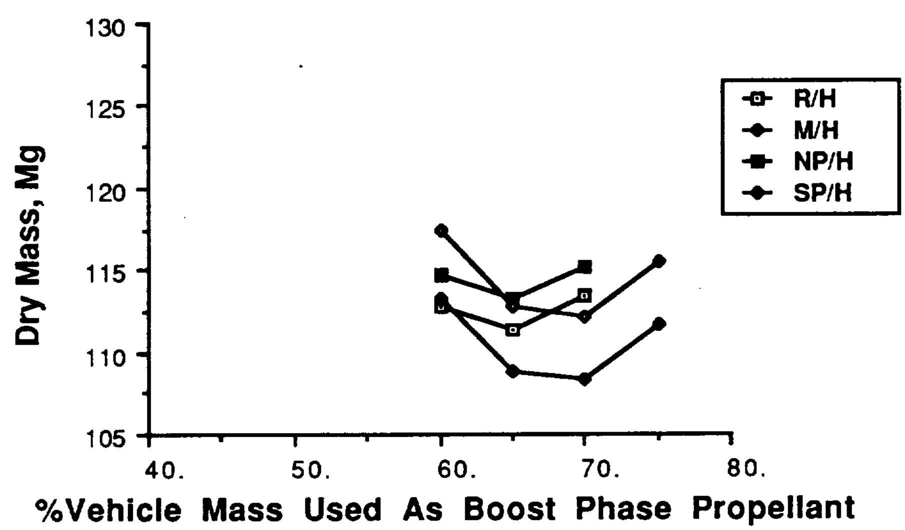


Figure 3.7-3 Dry Mass Versus Boost Duration for SSTO Using Translating Nozzles-Hydrogen Cooled Engines

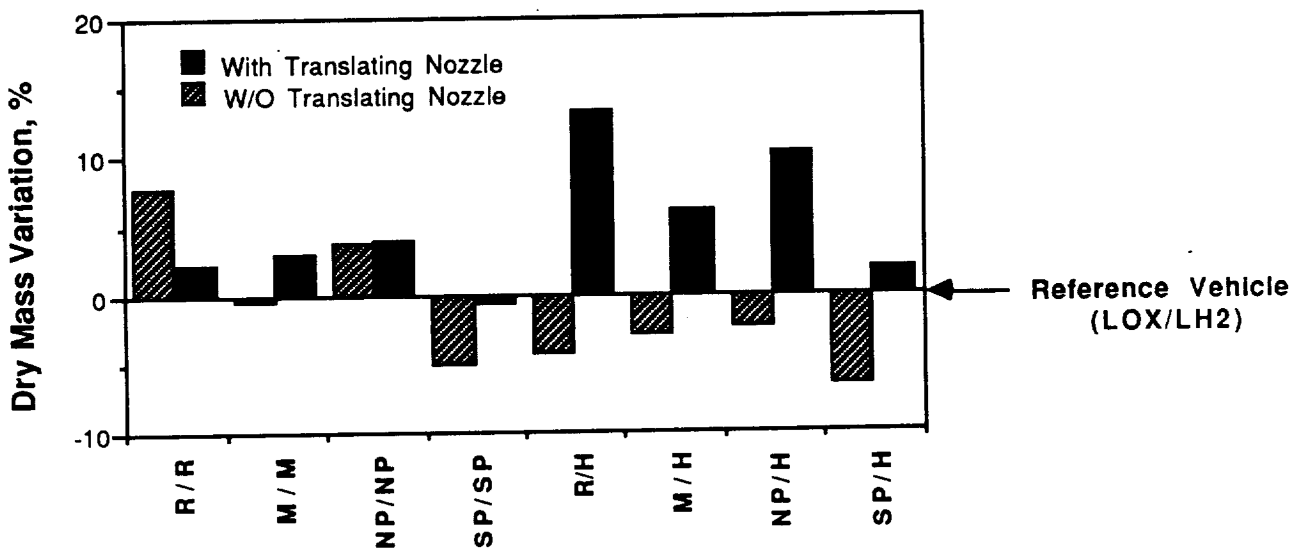


Figure 3.7-4 Comparison of SSTO Configurations With Translating Nozzle and Configurations w/o Translating Nozzle to Reference (LOX/LH2) Vehicle - Total Vehicle Dry Mass

The UFRCVs showed more sensitivity to the duration of the boost stage during which the BTNE operated at the low expansion ratio. Thus, Figures 3.7-5 through 3.7-7 show the curves generated for all nine engine options with the BTNE duration parameter represented by Bfrac values. These graphs show total vehicle dry mass values for varying boost stage duration as well. As for the SSTO, the minimum dry mass values from each graph was selected to represent the optimum configurations for this analysis. These optimum configurations are compared to the reference vehicle, and the optimum configurations that did not use the translating nozzle engines, in Figure 3.7-8 on a total vehicle dry mass basis. The results are similar to those for the SSTO. The use of translating nozzles on the engines with different fuel/coolant options resulted in vehicles with greater dry mass than for those vehicles where the translating nozzle was not employed and, usually, resulted in vehicles with greater dry mass than for the reference vehicle.

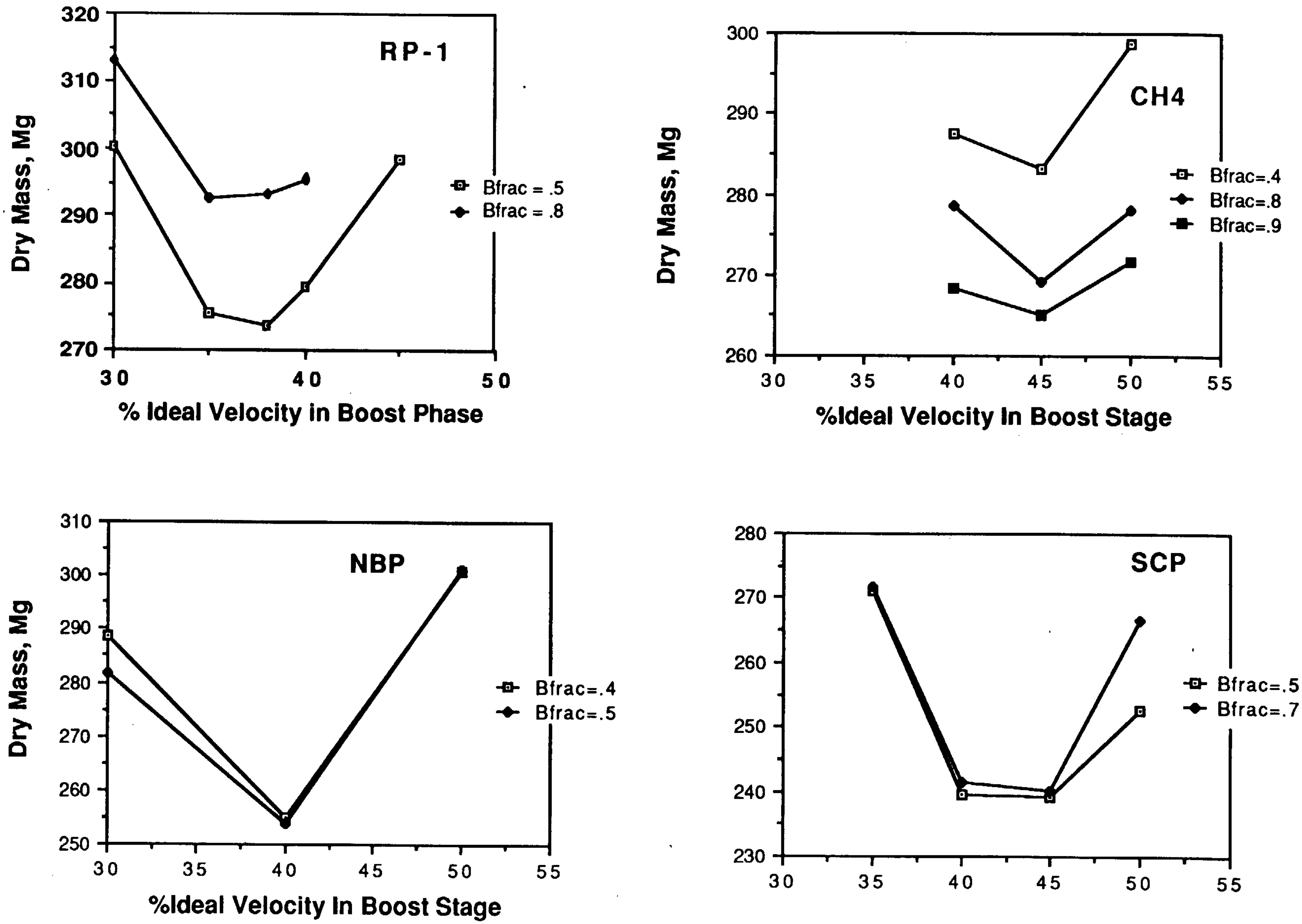


Figure 3.7-5 Total Vehicle Dry Mass for UFRCVs Using Translating Nozzles versus Boost Phase Duration-Fuel Cooled Engines

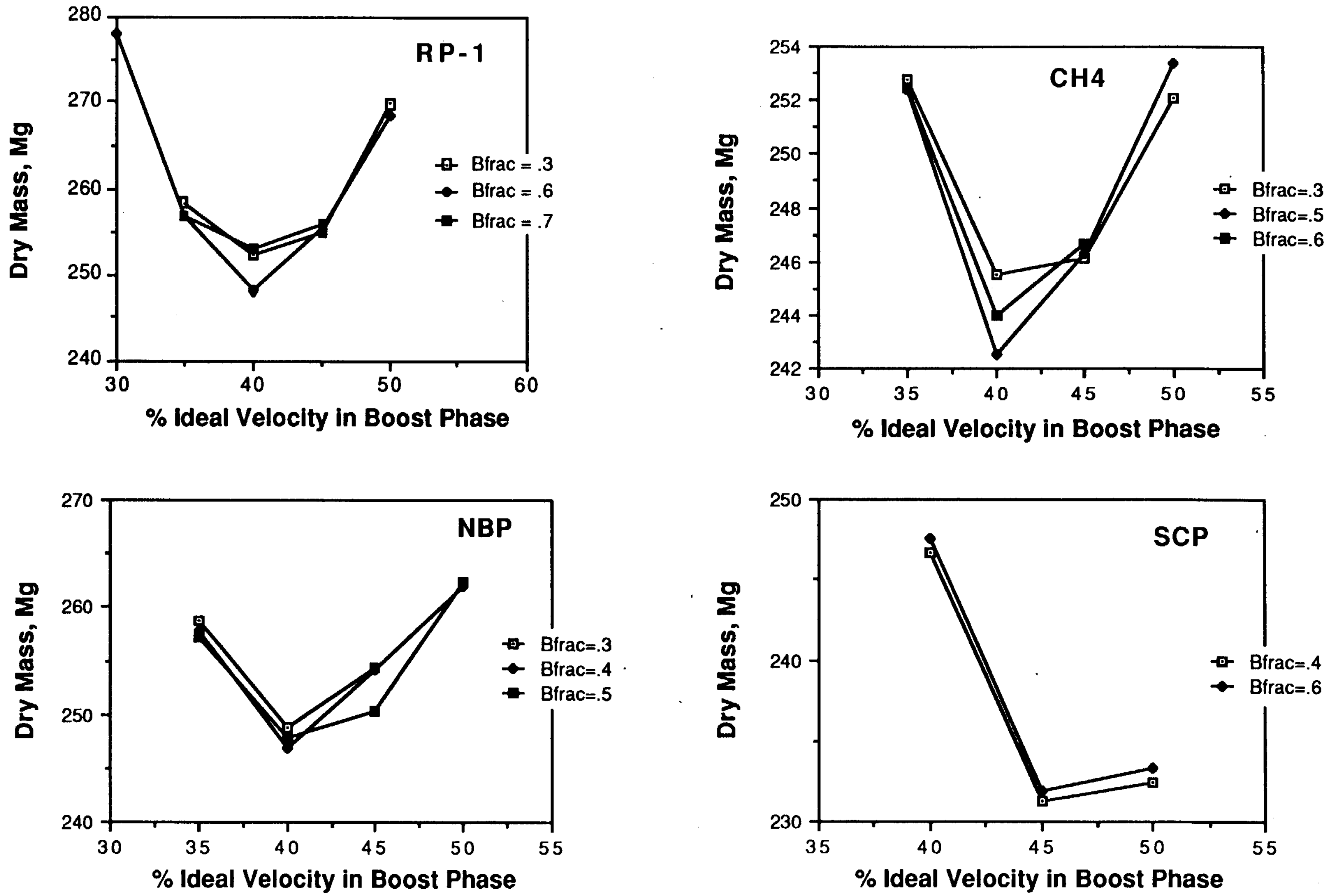


Figure 3.7-6 Total Vehicle Dry Mass for UFRCVs Using Translating Nozzles versus Boost Phase-Hydrogen Cooled Engines

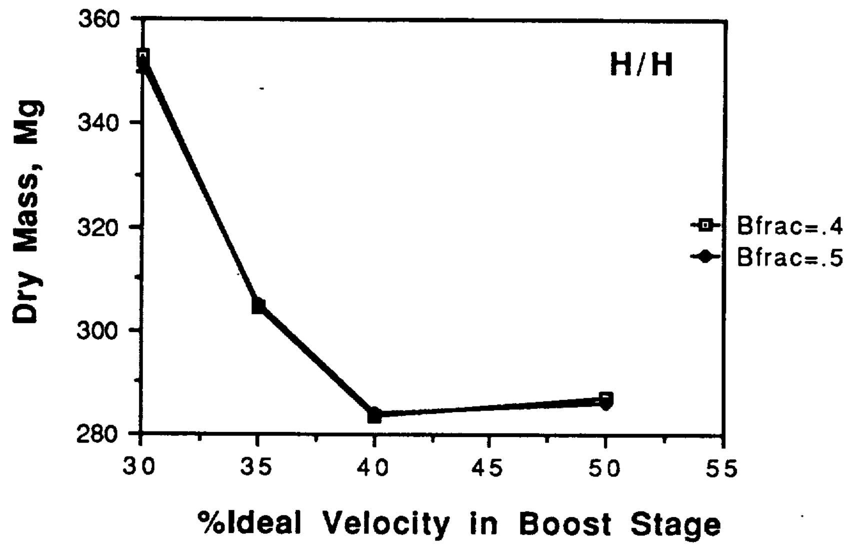


Figure 3.7-7 Total Vehicle Dry Mass for UFRCV Using Translating Nozzle Versus Boost Phase Duration-All Hydrogen Engine

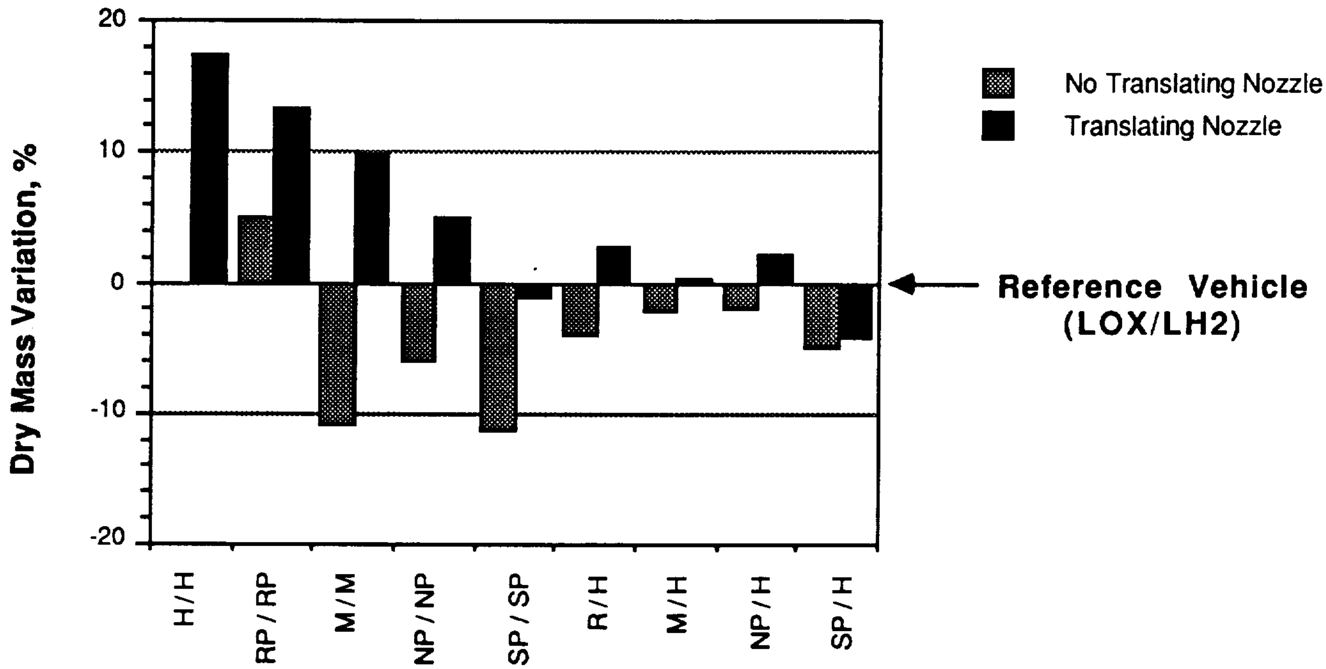


Figure 3.7-8 Comparison of UFRCV Configurations With Translating Nozzle and Configurations w/o Translating Nozzle to Reference (LOX/LH2) Vehicle - Total Vehicle Dry Mass

4.0 TASK 2.0

4.1 Objective and Summary

The objective of this task was to determine the preliminary design impacts of using subcooled propane versus NBP propane on a selected baseline vehicle and the ground support systems for the vehicle. Preliminary design options were to be generated for the bulk storage, distribution and thermal control of both NBP and subcooled propane for the ground support systems required to support a vehicle launch. Preliminary design options for pressurization and thermal control systems on the vehicle were to be established for both fuel options. All preliminary design options were to be generated at a level that would allow a rough order of magnitude (ROM) cost calculations to be conducted for the options. To complete the preliminary design impact for this task, the ROM costs were generated.

4.1.1 Task Breakout and Approach

This task was broken down into the subtasks shown in Figure 2.3-1. Both the ground support and vehicle systems followed the work flow depicted in Figure 4.1-1, which illustrates the method used to accomplish the objective. However, the vehicle analysis did not include any cost estimates for reasons discussed below.

Subtask 2.1 identified the subsystems necessary to support vehicle propellant requirements for the two fuel options and the appropriate design options for each subsystem. Selection of best alternative methods to achieve subsystem requirements were made for both propellant options.

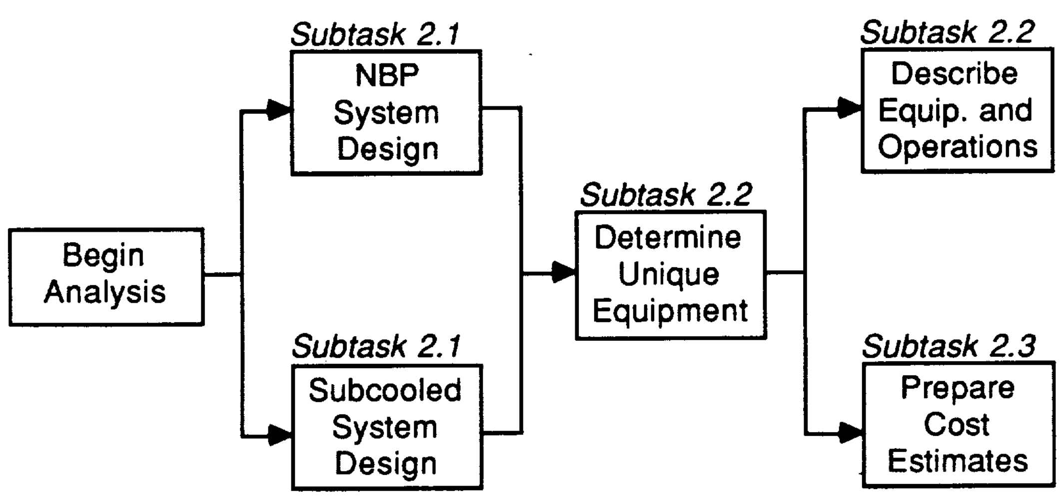


Figure 4.1-1 Work Flow for Task 2.0

Subtask 2.2 utilized the selected design options from Subtask 2.1 to identify specific equipment used to achieve the design options. This equipment was identified in sufficient detail to allow ROM cost estimates.

Subtask 2.3 used the identified equipment from Subtask 2.2 to calculate ROM cost estimates. Costs were estimated for equipment necessary for both fuel options. This was not done for the vehicle system.

The majority of the effort, including cost analysis, was focused on the ground support subsystems. It was believed that the source of major cost differences between using subcooled versus NBP propane would be the ground support systems. Previous analyses for only the vehicle indicated that ROM cost differences between subcooled propane and NBP propane fueled vehicles were negligible on a vehicle subsystem basis. Furthermore, internal company cost models lacked sufficient fidelity to accurately generate such small differences in costs. Thus, only the major differences in the relevant vehicle subsystems' designs for the use of subcooled and NBP propane were identified; ROM costs were not determined.

The interfaces between the vehicle and ground support were examined for impact on either system when the two fuels were used. Of particular initial concern was the probable requirement that the ground support system maintain the propellant in the vehicle in a subcooled state during launch hold.

4.1.2 Input Requirements and Ground Rules

Subcooled propane was assumed to be established at a temperature of 91.5°K. The appropriate physical properties of the two fuels used in the analyses are as shown in Table 4.1.-1. The vehicles used for ground support system analyses were the optimum subcooled and NBP propane fueled UFRCVs identified in Subtask 1.2. In order to magnify the ROM cost differences between the use of the two fuels specific design options, that would generate the greatest differences in cost, were selected from the identified alternatives .

Table 4.1-1 Propane Data Base

PROPERTY	COMMERCIAL	NBP	SUBCOOLED
Density (Mol/L)	11.32	13.18	16.48
P (MPa)	0.857	0.10135	7.0×10^{-4}
T (°K)	294°	231.04	91.5
Purity	95%	>98%	>98%
M (poise)	0.0373	.0685	2.399
H (J/Mol)	-1993	-8796	-21352

4.2 Ground Support System

4.2.1 Objective

The objective of this analysis was to identify the impacts to the ground support segment of utilizing subcooled propane (Sc) versus NBP propane. These impacts are further quantified in a preliminary ground support system design that allows for determining ROM costs. The particular subsystems examined included those providing for: achieving and maintaining the conditioned propellant; storage; transfer; distribution to the vehicle; offloading and system securing.

4.2.2 Approach

The approach was to develop a propellant ground handling system block diagram and then to analyze each element for impacts for the utilization of subcooled or NBP propane. The ground system block diagram design as shown in Figure 4.2-1 is based upon direct experience with the propellent handling systems at both Kennedy Space Center (KSC), FL and Vandenberg AFB (VAFB), CA. Alternative approaches for each element were developed and evaluated. Evaluation of alternatives was based initially on experience with further evaluations based on combining numerical analysis with experience.

The major subsystems addressed herein are; the delivery method for the basic commodity, the method by which the commodity can be conditioned to the proper temperature, the method for commodity storage and the method for transfer of the commodity to/from the vehicle. Additionally any special impacts which arise from the vehicle/ground interface (e.g. commodity thermal maintenance onboard the loaded vehicle) were evaluated. Thermal maintenance of the commodity is addressed in the storage section. The possible options analyzed for each subsystem and the issues associated with them are summarized in the trade study matrix, see Table 4.2-1. More detailed discussion of the options and the ones selected for each subsystem follow.

A few overall assumptions for the engine ground support system requirements were made. The baseline vehicle for this analysis is loaded in the vertical orientation and is placed in the launch position prior to propellant loading as is current spacecraft practice. The vehicle requires a loading of approximately 372,000 kgs. of either NBP or subcooled propane. This propane loading is accomplished in a 30 minute fast fill period (>90% of total required). Preconditioning (chilldown) and topping/replenish phases will occur outside this 30 minute period and the ground system will be capable of supporting these at approximately 1,893 liters per minute for a slow fill rate. These phases are customary to cryogenic propellant vehicles and mitigate such occurrences as nonuniform or excessive structural loading, insulation debonding or excessive pressure surges in fill lines and the vehicle.

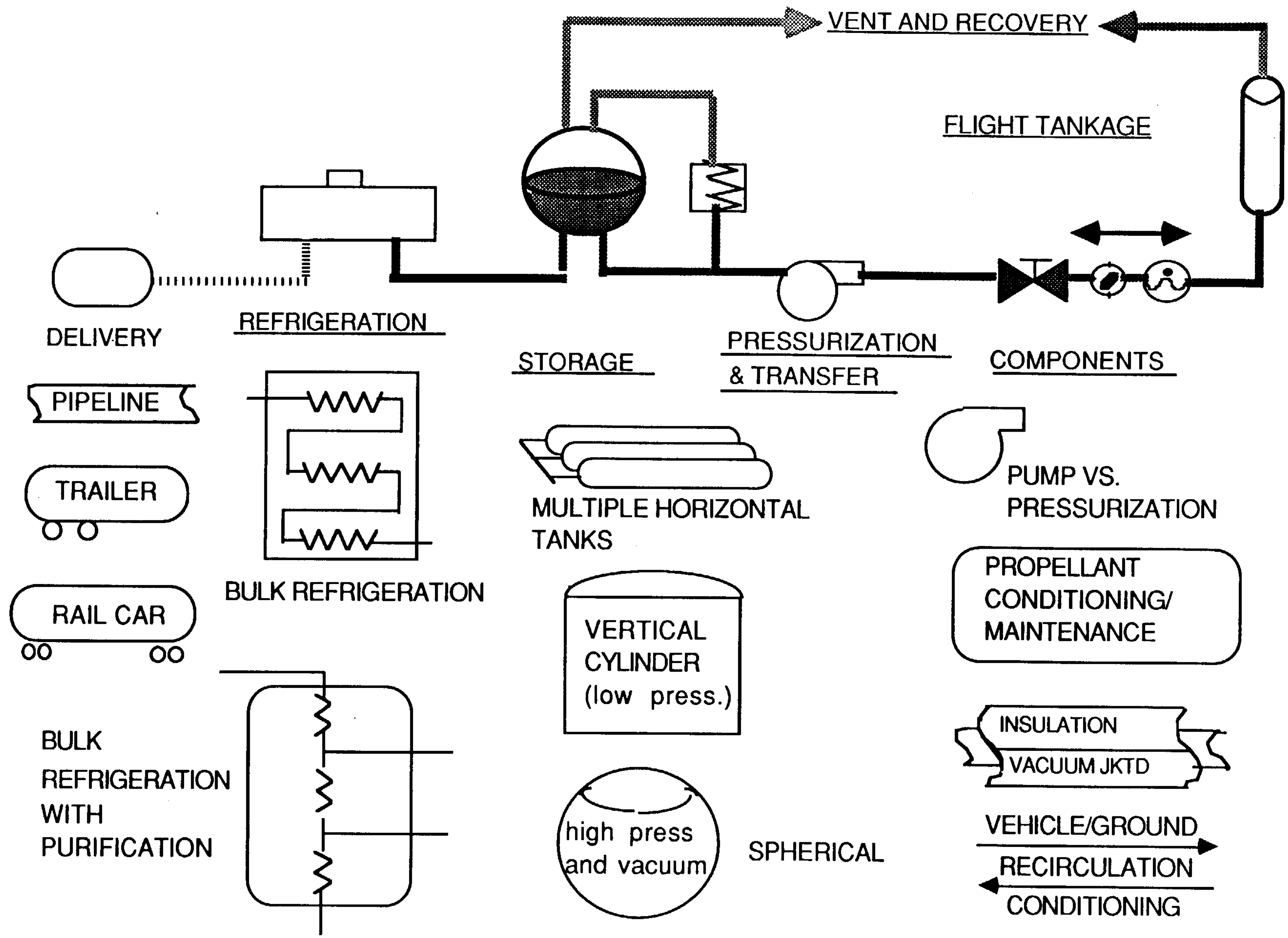


Figure 4.2-1 Propane System Block Diagram

Table 4.2-1 Trade Study Matrix

OPTION	USAGE	REMARKS
<u>DELIVERY</u>		
Pipeline	• For commercial grade only	• Due to purity requirement and quantity required suggest rail car to site
Railcar	• 114,000 liter chemical grades and commercial grade	• Tanker trailer is alternate for local movement • No present commercial capability to produce or distribute subcooled or NBP
Tanker Trailer	• 22,700 liter shipped under pressure ~86 MPa at 294°K	
<u>STORAGE</u>		
Shape		
Spherical	• Thermally more efficient • More costly construction • Vacuum & pressure capable	• Horizontal cylinders are general storage method for commercial grade propane
Cylindrical		
Horizontal	• Available in "standard" sizes • Generally smaller volumes • Vacuum & pressure capable	• NBP can be stored in horizontal or vertical cylinders--insulated. Vertical cylinder is favored for singular large quantity low pressure storage.
Vertical	• Available in larger sizes • Less costly construction • Low pressure only <.03 MPa • No vacuum capability	• Subcooled storage for large quantities (1.9×10^6 liters) favors spherical
Pressurization		
Autogenous	• Requires heat exchanger loop • Higher pressures & lower temperatures require more commodity and larger heat exchanger capacity	

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
<u>STORAGE (Cont'd)</u> Pressurization (Cont'd) Inert Gas	• GN2 less effective at lower commodity temperatures • GHe is more costly • Long term high pressure exposure can contaminate commodity	
Venting To Atmosphere	• Requires burner (disposal) capacity commensurate with increased quantity - large surges • Pollution issue	• Vent to atmosphere is recommended and required in all cases for emergency • For subcooled storage at less than ambient pressures vent system will require vacuum breaker and evacuation pumps
Recovery	• Recovered gas may be reliquified, distributed to other users, or disposed of at more uniform rate • Recovery system is more complex and costly (especially reliquification)	-
Number of Tanks Single	• More thermally efficient	• Commercial grade can be stored in multiple or single uninsulated horizontal cylinders
Multiple	• More readily available • More adaptable to changing requirements • More complex manifolding and operations	

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
STORAGE (Cont'd)		
Insulation Uninsulated	• For ambient storage only	• NBP for large quantities and stable requirements favors single vertical cylinder with foam insulation, <0.2% per day boiloff
Foam or Pressurized Annulus	• Most economical for temperatures above liquid nitrogen (77.6°K) • Foam is generally limited to smaller sizes • Typical tank would provide 1250 L/day boiloff (.07%)	• Subcooled requires spherical pressurized annulus for large quantity storage, <0.1% per day boiloff equivalent
Vacuum Jacketed	• Most thermally efficient • High cost	
REFRIGERATION		
Bulk Vaporization	• Requires Evaporization of 42% from .86 MPa, 294°K to NBP • Incapable to cool to subcooled	• Utilize refrigeration for NBP and subcooled
Refrigeration	• NBP requires 1.7×10^{11} Joules or ~7,000 J/Mol • Subcooled requires 5.4×10^{11} Joules or ~19,200 J/Mol	• Further study required for efficiency of refrigeration plus purification
Refrigeration plus Purification	• For subcooled temperatures most impurities will condense out • Might prove efficient in utilizing commercial grade	

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
<u>TRANSFER</u> Method *Pump	• Requires storage tank to provide pump NPSH only • Adds complexity to transfer system • Pump system reliability may require redundant pump circuits	• For 30 minute loading time typical system requires: For NBP 82.3 m head ~221 KW For Subcooled 189 m head ~510 KW
Multiple	• Separate rapid load and fine load pumps • Provides more flexibility for variable flowrates	
Singular	• Requires variable speed controller or pressurized chilldown/slowfill	• Separate rapid load and fine load pumps are recommended for both NBP and subcooled
Controller Fixed Speed	• More economical	
Variable Speed	• Allows good efficiencies for multiple flowrates • May be utilized by only one pump in multiple pump system	
*Pressurization	• Requires storage tank and pressurization system to supply head pressure for transfer • NBP requires 130 psig tank • Subcooled requires 325 psig tank	• NBP could utilize only multiple horizontal cylinder tankage • High pressure requirement for subcooled is very costly (nonefficient) for all storage tank types

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
<u>TRANSFER (Cont'd)</u> Line Size	• Increase in line size reduces friction loss and power requirements but increase environmental heating due to residency time • Subcooled requires more forcing power in a typical system application with comparable system heat input Ref. Table 4.2-3 • Neither subcooled nor NBP can tolerate slow fill flow-rates thru fast fill flow line size due to heat input	• Typical system with 152.4 m of line and 21.3 m elevational head is best suited with 25.4 cm diameter fast fill and 7.62 cm diameter slow fill line
System Piping Material	• Stainless steel is commonly used material for cryogenic temperatures and cleanliness considerations • Aluminum can be used for both NBP and subcooled however contamination is more of a problem	• Stainless steel (304L) is recommended for NBP and subcooled service

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
TRANSFER (Cont'd) Insulation Non-insulated {For 25.4 cm fast fill line with environmental heating only. Length of line is 150 m. Slow fill line size is 10.61 cm. Fast fill rate: 21,300 L/min Slow fill rate: 1900 L/min See also Table 4.2-3}	• Comparison and analysis indicates that for NBP: a) Fast fill flowrates initial temperature rise = 1.1°K after .25 cm frost formation = .7°K b) Slow fill flowrate initial temperature rise = 5.5°K after .25 cm frost formation = 3.3°K c) Slow fill in slow fill line initial temperature rise = 1.1°K after .25 cm frost formation = .61°K d) Temperature rises for subcooled are approx. 6 times NBP	• Non-insulated line is not practical for all weather service
	• Comparison with Space Shuttle ground servicing lines indicate for: a) Fast fill flowrates NBP temp rise = .55°K Subcooled temp rise = 1.0°K b) Slow fill flowrates in fast fill line NBP temp rise = 1.1°K Subcooled temp rise = 3.3°K c) Slow fill flowrate in slow fill line Subcooled temp rise = .8°K	• Insulated line system is recommended as most efficient for subcooled and NBP • Vacuum jacketed slow fill line may be required if temp rise is critical • NBP system may use only the fast fill line for both fast fill and slow fill

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
<u>TRANSFER (Cont'd)</u> Insulation (Cont'd) Vacuum Jacketed	• Most expensive • Analysis indicates for a) Fast fill flowrates NBP temp rise = .27°K Subcooled = .55°K b) Slow fill flowrate in fast fill NBP temp rise = .4°K Subcooled = .77°K c) Slow fill rate in slow fill line NBP temp rise = .33°K Subcooled = .66°K * Included in temp rise is pump effect	
Components	• Valves used for subcooled service must be extended bonnet type • Seals used in subcooled system must be designed for cryogenic temperatures	• Components for subcooled service will be cryogenic designs while NBP may use ambient service in most cases
<u>INSTRUMENTATION</u>	• Sensors must be of cryogenic range for subcooled • Lines between instrument and system must utilize cryogenic design practices • For subcooled tanks, areas, and lines which may operate at vacuum - 0_2 detection devices will be required	• Subcooled service instrumentation will be special service and more complex design. Typical increase in cost is 1.5 - 5 times ambient service instrumentation

Table 4.2-1 Trade Study Matrix (Continued)

OPTION	USAGE	REMARKS
<u>VEHICLE INTERACTIONS</u>		
Geysering	• Subcooled propane is transferred well below saturation temperature and has little possibility to form geyser • NBP has geyser potential equivalent of STS LO2 depending on vertical rise to vehicle tankage	• NBP system and operations must mitigate geysering potential
Vehicle Propellant Conditioning	• NBP can be conditioned by venting to atmosphere • Subcooled conditioning requires refrigeration system or large capacity evacuation system to evacuate the vehicle tank ullage	• Further study required for vehicle propellant conditioning
Vehicle Vent	• For large capacity venting of vehicle tank a separate (from ground system vent) vent system is recommended	• Requires separate vehicle vent for both NBP and subcooled propane systems

4.2.3 Discussion of Analysis and Results

4.2.3.1 DELIVERY

Various purity grades of propane are available from commercial sources. The standard commercial grade is available thru numerous pipeline networks. This commercial grade has a minimum purity of 95% and most of the delivered product is close to 98% pure. However, the experience of the rocket engine manufacturer, Aerojet Tech Systems expressed via telecon with the author is that the remaining impurities in the commercial grade will cause excessive 'coking' during rocket engine operation and contamination of the ground propellant distribution system. The recommendation is to use the chemical (or aerosol) grade which has a minimum purity of 98%. Additional specifications for this chemical grade are:

- No sulfur as established by the copper strip test ASTM D1838-84
- No moisture
- No N 2
- Ethane <.2% mole
- Isobutane <2.0%
- 97.9% minimum saturates

Chemical grade is not available thru pipelines and must be delivered by rail car (113,550 L.) or road tanker (22,710 L) trailer. Chemical grade is delivered at ambient temperature and at the saturation pressure of approximately 0.857 MPa. There is no commercial supply of chemical grade propane in either the NBP or Sc states; although in these states could be transported using equipment similar to that for liquid nitrogen or liquid oxygen. This equipment would provide adequate insulation for Sc and NBP propane. The NBP could be cooled and shipped in the manner of LN2 and L02 (22,710 - 30,280 L) without venting the trailer during shipment. The heat load into the trailer would cause some pressure rise and temperature increase which could be reconditioned by venting to a flare system (or controlled atmospheric vent) at the receiving station. The transportation of subcooled propane would necessitate the addition of an evacuation system, designed to maintain near vacuum conditions, to keep the propane conditioned at the low saturation pressures (10^{-8} MPa). Also, an O2 monitoring system would need to be added to the trailer system to detect hazardous leakage into the evacuated storage. These additions would be costly and dictate unique trailers for only this usage and is not cost effective for subcooled propane supply. Depending on commercial incentive the commercial supply of NBP propane may be possible. However, for the remainder of this analysis it was assumed that the propane would be delivered under ambient conditions.

4.2.3.2 STORAGE

Options for basic storage solutions are numerous and are tabulated in Table 4.2-2

Most combinations are available commercially although several combinations are much more common (e.g. large vacuum jacketed tankages are singular and spherical and uninsulated tanks are most common cylindrical and can be manifolded together). In order to evaluate the alternatives the quantity of propellant to be stored and the conditions for storage must be determined. The vehicle capacity of approximately 372,090 kg. is 638,700 L of NBP or 510,700 L of Sc. propane. A typical loading of the space shuttle uses 757,000 - 795,000 L of L02 to supply a vehicle with 549,000 L. Part of the loss is boiloff for childown of the facility and vehicle prior to loading, part is to maintain the vehicle propellant condition and much (79,500 - 170,000 L) is used for engine conditioning. Cryogenic storage tanks are seldom drained to below 20% capacity in order to prevent thermally cycling the tank. The VAFB ground support system (GSS) experience is that allowance must be made for additional growth quantity. The VAFB tank was sized prior to an increase in the engine conditioning flow and its 1,135,000 L capacity is now marginal for a load-drain-reload scenario. It is noted here that the commitment to a storage tank size is done very early in the program phase and is sensitive to downstream changes in requirements, either in vehicle or operational demands. The tankage for subcooled propane will be more thermally sensitive and thus require more complex tankage and will, as a result be less adaptable (more costly) to change. Based upon this background the storage tank is sized at 1,900,000 L for both NBP and subcooled.

Table 4.2-2 Storage Option Matrix

Number of Tanks	Single	Multiple		
Shape	Spherical	Vertical Cylinder	Horizontal Cylinder	
Insulation	Uninsulated	Insulated Non-Double Wall	Insulated Double Wall	Vacuum Jacketed
Pressurization	Autogenous	Inert Gas GN2/GHE		
Venting	To Atmosphere (Flare)	Recovery/ Reliquification		

In considering the number and type of tanks required to contain the necessary volume, multiple tankage systems of smaller sized tanks are more complex to operate and are less thermally efficient. However they are cost effective if the thermal requirements can be met with 'off the shelf' tanks. Tankage in excess of approximately 190,000 L must be of the vertical cylinder or spherical configuration in order to use 'off the shelf' designs.

For an uninsulated NBP, spherical propane tank of a volume of 1,900,000 L, the most thermally efficient tank design would result in a boiloff of 16% of the volume per day. For an acceptable boiloff in the range of <0.2% day a single wall insulation of approximately .3048 m would be required. However insulation of this thickness is easily damaged and does not weather well (Per KSC, NSTL and VAFB experience). A double wall insulated annulus purged with an inert gas can provide adequate insulation and is the next cost increment solution. A common tankage type to utilize this insulation method is the vertical cylinder tank with a flat bottom and domed top. Standard Union Carbide designs range from 500,000 L to over 4 million L with a typical height to diameter ratio about unity. A tank of this configuration and typical performance would generate a boiloff of 650 L/day. The vertical cylinder design is a low pressure <0.035 MPa only design. If higher pressure is required for pump suction head or pressurized transfer then a spherical design must be utilized.

Subcooled propane requires a tank design which can withstand a high vacuum due to the saturation pressure at 91.5°K being close to the triple point pressure at 85.9°K of 3.0×10^{-10} MPa. The typical vertical cylinder will not withstand the vacuum and thus the next choice is a spherical tank design. This design is used for L02 storage at both KSC and VAFB. Using the thermal performance of those tanks a 1.9 million L spherical tank would generate a daily boiloff of 1250 L/day for subcooled propane. Boiloff (evaporation) would only occur if the tank ullage pressure were kept at the saturation pressure. If not constantly evacuated the heat input to the tank would result in a .16°K per day rise in the bulk temperature. It is conceivable that the evacuation system could thus be used only intermittently. For either the spherical or vertical cylinder tanks the insulating material is usually perlite but other common insulations can be used. Since the subcooled and NBP temperatures are above LN2 temp. of 77.6°K gaseous nitrogen can be used as the purge gas. A vacuum jacketed tank, rather than an insulated one, would provide increased thermal performance but at a diminished cost return. A typical vacuum jacketed tank would have a boiloff rate of only 26 L/day of subcooled propane.

Ancillary to the tankage size and type are the vent and pressurization systems. The venting boiloff can be either disposed of to the atmosphere or contained and reliquified. Pressurization systems can be either autogenous or via inert gas, GHe or GH2. Considering the complexity and high cost of reliquification systems for which an emergency atmospheric vent would be required it was determined that venting be done to the atmosphere. Various types of of disposal mechanisms are available such as flare stacks and catalytic burners. Each requires proper environmental permits. The flare system is the most common and reliable. These systems must be sized for storage tank ventdown as might occur after a loading operation when the tank had been pressurized.

4.2.3.3 REFRIGERATION

Unless NBP propane is made available both NBP and subcooled propane must be cooled in the vicinity of the launch facility. Bulk evaporation and mechanical refrigeration are possibilities for conditioning the propane. In order to condition propane delivered at 0.857 MPa and 294°K to the NBP of 231°K a thermal extraction of approximately 6,983 J/Mol is required. For the evaporation process the heat of vaporization is utilized to provide the refrigeration. At an average of 16,915 J/Mol the process requires the boiloff of 42% of the original quantity to achieve the NBP state. Although a recovery-reliquification system could be added, this process is inefficient compared to mechanical refrigeration.

It is not possible to use the evaporation method to achieve the subcooled state. To achieve the subcooled state mechanical refrigeration must be used. The requirement is for 19,273 J/Mol of refrigeration. The complexity and expense of the refrigeration system increases geometrically as the final temperature decreases. In this case 231°K versus 91.5°K, for NBP and subcooled respectively, causes a very large increase in plant cost. Of large concern again is the reduced pressure at which subcooled propane must be handled. Leakage of air (O2) into (and thus not easily detectable) the propane would produce a hazardous shock sensitive mixture. Experience shows that a leak out of a system, as in the case of a NBP system, is easier to detect and has more commonly available design solutions than a leak into a system.

A possibility unique to the subcooled propane refrigeration is that commercial grade could be procured and purification could occur during refrigeration. Most of the contaminants, including the heavier (higher degree of coking) hydrocarbons , will condense out during refrigeration. Methane will freeze at 90.7°K which is above propanes freezing point of 85.5°K. Nitrogen will remain and must be removed by other means if it is present in the raw stock. Conversation with Air Products representatives indicates that this is a favorable possibility for subcooled propane production.

The rate of refrigeration is dependant upon several factors. The launch rate, the delivery rate and the method of refrigeration will all affect the rate to be required. Unloading of tankers to large capacity storage is generally accomplished in waves of tankers to minimize the system chilldown losses and for better crew efficiency. Each tanker is generally offloaded at approximately 3,800 L per minute. For refrigeration at tanker offload this would require 1.013 x 10^9 J/min for NBP and 3.586 x 10^9 J/min for subcooled. Assuming a launch rate of 2/month with a ten day period available for propellant conditioning the requirements are reduced to 1.48 x 10^7 J/min for NBP and 4.01 x 10^7 J/min. This rate is based on 8 hours of operation per day. The refrigeration could proceed 24 hours per day at a reduced requirement.

4.2.3.4 TRANSFER

In order to supply the 372,000 kg. of propellant to the vehicle in 30 minutes during the fast fill phase, flowrates of 21,300 L/min at NBP or 17,000 L/min at subcooled condition are required. A loading system design has been developed utilizing similarities to Apollo and Space Transportation System (STS). It was assumed that the storage tanks are connected to the vehicle by 1600 meters of transfer line which has a vertical rise of 230 meters to the vehicle interface. The actual configuration will depend on site characteristics but these values present realistic relationships of the thermodynamic data. The flow of both NBP and subcooled propane was analyzed through lines varying from 15.24 cm to 40.64 cm in diameter and through insulated or bare line. The data is summarized in Table 4.2-3. Head loss is comprised of both elevational and frictional. The frictional head loss is made up of line loss and component losses. The increase in head loss of transferring subcooled propane, versus NBP propane, is due to its increased viscosity and density. Heat input into each system is comprised of environmental heating, frictional heating and, in the case of a pumping system, the heat input due to pump inefficiency. Values for environmental heating thru the insulation system are based on experience with a similar line of the VAFB STS LO2 system. Values for component head loss are based on those actual C_v values tested for the KSC STS LH2 system and the VAFB LO2 system. The pressure drop attributable to the line friction loss is calculated using the standard Darcy equation from L/D values for the assumed transfer lines. The pressure drop for system valves was adjusted for the increased viscosity of subcooled propane.

Table 4.2-3 Summary of Transfer Flow Analysis

	Line Diameter, cm				
Q = 12,402 kg/min	15.25	20.32	25.4	30.5	40.64
Head Loss m					
Elevational	21.33	21.33	21.33	21.33	21.33
Frictional					
Subcooled	898.5	216	67	28.6	12.2
NBP	503	119	36.6	15.2	6.1
KW Pump Required					
Subcooled	2,477	639	238	134	74.6
NBP	1,413	379	156	98	44.7
Heat Input J/mol <Δ°K> Insulated Subcooled	808 <8.3>	225 <2.5>	105 <1.1>	78 <.83>	82 <.94>
Non-insulated (frosted) (Δ°K)					
Subcooled	875 <9.4>	318 <3.4>	215 <2.4>	215 <2.3>	266 <3.0>
NBP	471 <2.6>	155 <.83>	100 <.55>	87 <.50>	115 <.61>

KW required for pumping is calculated by

$$\text{Kilowatts} = \frac{\text{LPM x Head Loss x Sp. Gr}}{3406 \times \text{EFF}}$$

where pump efficiency in all cases is selected at 75%; a value at which both the KSC LO2 and VAFB LO2 pumps operate.

A line size of at least 20 cm is required for transfer of NBP propane due to its pressure drop. The 20 cm line is a marginal choice based on the pump KW required. When heat input effects are reviewed, the longer residency time for a 30.5 cm line tend to balance out the increased frictional heating of the 25.4 cm line, thus the 25.4 cm line size is recommended. The approximately 0.55°K temperature rise for the uninsulated line would be acceptable. However, it must be noted that windy or rainy conditions will increase the heat input of a frosted line by about 10 times which would be a 4°K to 5.5°K rise and is not considered acceptable. Thus, the insulation system chosen for this line needs to primarily provide a weather barrier.

Subcooled propane with its increased head loss requires at least a 25.4 cm diameter transfer line. The pump KW required is less than (about 1/2) that required for the VAFB STS LO2 system due to the lower density. Again, the longer residency time for the 30.5 cm system does not make this choice cost effective. The reduction in pump KW required should be considered in a costing trade study. The insulation system as used to develop these data is a 12 cm tempmat plus 65 cm foamglass covered with a protective fiberglass barrier. Vacuum jacketed piping would provide improved performance but its increase in installed cost is approximately 10 times the insulated system and is not cost effective.

The choice of material for the piping is stainless steel for both NBP and subcooled. Although aluminum could be used it is less common and does not provide significant cost savings.

The high head requirements for transfer of both NBP and subcooled propane prohibit the use of a low pressure (vertical cylinder) storage tank in a pressure fed transfer system. The increased cost of tankage capable of supplying the required transfer pressure is greater than the cost of a pumping system. Thus it is recommended that a pump system be used for both subcooled and NBP propane. Although pump systems are quite reliable, as experienced by the KSC Apollo and STS systems, a redundant pump-motor-controller is commonly provided and is recommended. A pumping system would allow the use of a vertical-cylinder tank for NBP storage and moderate pressure spherical tank for subcooled propane storage.

If the head requirements for vehicle chilldown and topping are within the pressure capability of the storage tank (in the case of the spherical tank) then a fine load, also known as slow fill, pumping system may be eliminated.

Variable speed pumps do give a great deal of versatility to a pumping system and should be considered especially on the slow fill system. The required pump turn down ratio of 10:1 is greater than the general range for acceptable variable speed pump performance however. Note that there are no requirements peculiar to NBP or subcooled propane for variable speed pump systems.

For the slow fill conditions of 1900 L/min the option of using the fast fill line versus a separate slow fill line was analyzed. The results are tabulated in Table 4.2-4.

Table 4.2-4 Slow Fill Line Usage Comparison

	Temperature Rise	
	NBP	Subcooled
Using Fast Fill Line	2.28°K	3.6°K
Using Slow Fill Line	.83°K	1.5°K
Values do not include pump Δ T of .55°K for Sc and .33°K for NBP		

The temperature rise for the NBP condition using the fast fill line may be allowable. This would eliminate the slow fill line in that system. However, the replenish valve would still be needed in that system.

An area related to pumping and requiring a different system design for subcooled propane transfer is the tendency for the transfer system to attain the saturation pressure (vacuum for Sc propane) if the pressure source (pump) is shut down. This requires that the seal designs provide for sealing against the vacuum. This will effect the design of quick disconnects to a great extent. Designs are available commercially but they do increase component cost. Due to this potential for vacuum, and an inleak of oxygen as mentioned in the storage section, additional oxygen monitoring of the system internals will be required.

Component design differences between the subcooled and NBP systems will be the same as those between cryogenic service equipment and ambient service equipment. Valves will require extended bonnets for subcooled propane service. Component soft goods for ambient service are generally rated to only 244°K and thus cryogenic compatible softgoods will be required for both NBP and subcooled propane systems.

Instrumentation will also reflect a cryogenic versus an ambient service rating. Transducer and gage line routings are more critical with subcooled service to both protect the instrument and prevent heat leak into the system.

4.2.3.5 VEHICLE-GROUND INTERACTIONS

For the Space Shuttle loading with LO2 propellant geysering is of concern. This condition occurs when the liquid rising in the vehicle feedline loses the elevational potential (pressure) and thus becomes superheated for that lower pressure. The fluid then boils and the gases form a 'taylor' bubble which acts as piston driving up the propellant in the feedline. The resultant geyser creates sloshing in the vehicle and possible ullage pressure collapse and a pressure surge in the voided feedline. This condition is possible for NBP due to the propellant condition being close to saturation conditions. The occurrence is unlikely for subcooled propane due to its great amount of subcooling when transferred under pressure.

Subsequent to loading it is desirable to maintain the desired vehicle propellant conditioning. For NBP this can be accomplished by venting to atmospheric pressure and maintaining a replenishing flow. For subcooled conditioning two options were considered. Either the vehicle must be maintained at the saturation pressure (evacuated) or refrigeration must be provided to balance the vehicle heat load. Evacuation of the vehicle tank is undesirable as it again adds the complication of hazardous leakage into the tankage and the addition of leak detection systems. Refrigeration can be provided by a heat exchanger or via propellant recirculation. Comparison with the Space Shuttle external tank heat load gives a vehicle heat load of 27.5×10^6 J/min. Recirculation flow can only provide 92.2 J/mol for every one degree K difference between the recirculated propellant temperature and the vehicle propellant temperature. For a 2.7°K differential and 100% mixing efficiency the required recirculation flow would be 6,400 L/min. This flow rate is not practical.

Evacuation of the ullage utilizing the heat of vaporization requires a evacuation rate of 50 kg/min. If not vented, the vehicle heat load will result in a 1°K temperature rise every 27 minutes. Further study is recommended to investigate possible heat exchanger methods versus evacuation of the ullage.

If the vehicle operating scenario requires a rapid ullage depressurization then a separate vent (flare) system is recommended for the ground system. This is common with the STS GSS systems.

4.2.3.6 GROUND OPERATIONS SCENARIO

The major characteristics of each subsystem and the differences for using subcooled versus NBP propane are summarized in Table 4.2-5. The overall ground operations scenarios for each are similar. The differences are in magnitudes or auxiliary system requirements (e.g. ullage evacuation and hazardous leak detection) and that the subcooled propane system may utilize commercial grade stock.

Table 4.2-5 Impact of Subcooled vs NBP Propane on Subsytsems

Delivery	Refrigeration	Storage	XFR	Vehicle
- Chem Grade Tanker Manual offload * Sc may use commercial with purification	- Begin 10 days prior to load * Sc requires greater capacity * Sc may purify * Sc requires vacuum system	- vert cyl . for NBP - Flare stacks * Sc - spherical tank	- ~ same line size - R/L & F/L lines * Sc requires more instrumentation * Sc requires more component design * Sc leak detection/ 02 monitoring * Sc pump requires more KW * Sc lines better insulation (might be a wash)	- Separate vehicle vent * Sc requires propellant conditioning system NBP needs geyser system

*= where Sc is different

- = where NBP and Sc are similar

Operations will begin with tanker offloading. This procedure will be as much local manual control as possible. This requires one remote operator to monitor storage tank conditions and system conditions. The unloading operation will involve 3 tank trailers or one rail car at a time. Flowrates will be about 11,400 L/min total. For the subcooled system, the storage tank ullage may require pressurization to steady the tanker transfer offload flow. When the storage tank filling operations are complete, the refrigeration process can begin. Although refrigeration can take place between waves of trailers it would not normally be done until resupply was completed. Resupply will take approximately 6 days at 6 trailers/day. STS experience indicates that 2 waves of 3 tankers can be offloaded in an 8 hour shift. The refrigeration process will be an automated procedure with remote monitoring and control. The refrigeration systems have been designed to provide the required refrigeration in 5 to 10 days (11 or 8 hours/day of operation). Maintenance of propellant condition within the storage tank during long hold periods will require periodic refrigeration for the subcooled propane system and venting of the NBP system.

Propellant transfer will be accomplished using separate fast fill pump/transfer line system and slow fill/transfer line system. The procedural steps for fill will be: facility chilldown; vehicle chilldown; slow fill to establish liquid in the vehicle to a point at which a stable rapid fill procedure can be accomplished, rapid fill to approximately 98% full, decrease of fill rate to slow fill to 100% and then maintenance of that 100% level. The subcooled system will require more pump power to achieve the same flowrates as for NBP propane.

As mentioned previously the major operational differences for the subcooled system will be: (1) The tendency for the system to form a vacuum if the pressurization system fails, and (2) the difficulty of maintaining vehicle conditioning during pad hold periods. Dynamic affects such as water hammer will be present on both systems and increased for the subcooled system due to its increased density. Vehicle offloading will be similarly handled for each system by pressurizing the vehicle ullage and depressurizing the storage tank ullage. As in the case of tanker offload the subcooled system storage tank may require being kept at above ambient pressure in order to stabilize the offload. All fill/drain operations will be remotely controlled/monitored and will be automated. System securing will be through warming and inert purging of the transfer lines. For the subcooled propane system an initial and periodic vacuum leak checking of the transfer system will be required.

4.2.3.7 COST ANALYSIS

Rough order of magnitude (ROM) costs were estimated by contacting various providers of equipment, such as tanks, pumps, refrigeration units etc. and NASA facilities offices, which had previously bought such equipment. The costs were obtained for five major areas: 1) Commodity cost, 2) Storage tankage, 3) Refrigeration system, 4) Transfer system and 5) Evacuation and reliquification. The cost of the latter system is based upon an assumption of using ullage evacuation to maintain the propellant condition in the vehicle during launch hold. Although this may not be the preferred method, see vehicle impact discussion, the cost for evacuation system should be comparable to possible alternates. Summary of the costs for the five areas are shown in Table 4.2-6.

Propane delivery costs are the same for both types of propane. However, additional losses for transfer system and vehicle chilldown, estimated at 114,000 L per loading cycle based on STS experience, for subcooled propane adds $15,000 per launch to the delivery costs. The propane delivered is aerosol, or chemical, grade, (98 percent purity).

Storage costs assume a vertical cylinder tank for NBP propane. This design is the most economical but does require a pumping system to transfer the propane to the vehicle. Spherical tanks are used for subcooled propane. Two alternates exist, both of similar design. The higher pressure tank has a thicker inner wall and allows the transfer of subcooled propane by pressure rather than using a pump system; but it also requires a larger vaporizer capacity which is reflected in the cost for this tank. The lower pressure tank is lighter, but requires a pump system for transfer. All the tanks have stainless steel inner walls and carbon steel outer walls.

Refrigeration is required to chill the delivered propane to the conditions desired for storage. The added cost for the refrigeration needed to obtain subcooled conditions is six times greater than that needed for the NBP propane. Provided for comparison is the refrigeration cost of obtaining subcooled propane in five days, the ten day period is preferred. Of interest is that the refrigeration process can be used as a purification process as well. This may allow purchase of commercial grade propane, rather than aerosol grade. Then, as refrigeration occurs, the commercial grade propane can be purified to the proper extent. This has a potential savings of $20,000 to $40,000 per propane delivery.

Pumps required to transfer the two types of propane include variable speed controllers. Each pumping system consists of a slow fill and rapid fill pump. No redundancy in pumps is assumed. The variable speed capability provides more operational flexibility at comparable costs to multiple fixed speed pumps.

Transfer piping consists of 150 m each of 10.16 cm and 25.4 cm nominal diameter schedule 10 stainless steel piping. As described above in the ground support system analysis, the insulation method for NBP propane is a 1.27 cm layer of fiberglass or pour foam with a weather shield. The insulation for the subcooled propane is 10.16 cm of pour foam or a buildup of fiberglass mat and foam glass. The listed piping costs include installation.

The costs are added by summing all the values under the NBP propane column to obtain a total of $2,399,000. The costs for subcooled propane has two values, depending upon whether pumped or pressure transfer method is used. Summing the values in the subcooled propane column for cost categories one, three, five and piping in four, then the high pressure tank costs, or the low pressure tank costs and the pumping system costs are finally added to obtain two different totals. The smallest value is $7,334,000 while the larger value is $7,339,000. Both values are increased by $15,000 per propane delivery.

4.2.4 Conclusions

The impact on the ground systems in various subsystem areas is described in Table 4.2-5 and the respective costs are shown in Table 4.2-6. Although the cost difference between using NBP or subcooled propane is approximately $5,000,000, this value is very small when compared to the cost necessary to build a new launch facility or new launch vehicle. The only significant design issue not fully resolved is the propellant conditioning in the vehicle while on the ground. Although the cost of the system necessary to maintain conditioning is low, the specific design needs to be integrated with the vehicle flight system used for propellant conditioning for greatest efficiency.

Table 4.2-6 Cost Analysis Summary

ITEM	COST $1,000 NBP	COST $1,000 SUBCOOLED
1. AEROSOL GRADE PROPANE		
1.5 MILLION LITERS	220	220
DELIVERY COST	64	64
ADDITIONAL CHILLDOWN LOSS		15 PER LAUNCH
2. STORAGE TANKS		
.7 MPa INSULATED	1,100	
.7 MPa DOUBLE WALL		2,600 (No Pump)
.28 MPa DOUBLE WALL		2,100 (With Pump)
3. REFRIGERATION		
AMBIENT TO 231°K (10 DAYS)	500	
AMBIENT TO 91.5°K (10 DAYS)		3,000 (6,000 - 5 days)
4. TRANSFER SYSTEM		
PUMP, MOTOR, CONTROLLER		
NBP 1,250-12,500 Kg/MIN	199	
NBP 0-2,000 Kg/MIN	91	
Sc 1,250-12,500 Kg/MIN		373
Sc 0-2,000 Kg?MIN		122
PIPING AND COMPONENTS		
10.1 cm DIA. 152 M 1.3 cm INSLN	40	
25.4 cm DIA. 152 M 1.3 cm INSLN.	185	
10.1 cm DIA. 152 M 10.1 cm INSLN		195
25.4 cm DIA. 152 M 10.1 cm INSLN		760
5. EVACUATION		
VEHICLE MAINTENANCE		500
	$2,399	$7,334+15 p/LAUNCH

4.3 Vehicle System

4.3.1 Objective

The objective of this analysis was to determine the significant impacts on vehicle design of using subcooled versus NBP propane as a fuel in the boost stage of the UFRCV. As discussed above, this impact analysis did not include ROM cost calculations.

4.3.2 Approach

4.3.2.1. Summary

The optimum UFRCVs determined in Subtask 1.2 for subcooled and NBP propane were used as the basic vehicles. Fuel cooling was assumed. Subsystems of the entire vehicle were examined in order to identify those key subsystems that would be affected the most by the use of subcooled versus NBP propane. After identification of these subsystems, various design options to achieve the subsystem requirements were determined based upon internal experience and some limited numerical analysis. From these design options, a specific one for each subsystem is selected. Since ROM costs were not to be calculated, an identification of the detailed equipment for the specific subsystem design option was not prepared. However, specific issues associated with identifying this equipment have been determined.

4.3.2.2 Ground Rules and Assumptions

The boost stage of the two stage vehicles with fuel cooling was evaluated. A heat input into each propane tank of 11.7 MJ/min was used to evaluated propellant conditioning options, this is based on the maximum heat leak allowable (665,000 BTU/hr) for the STS LO2 tank[13]. Slow fill and fast fill flowrates of 1900 L/min and 17000 L/min were assumed based upon the ground support system analysis. During the loading of subcooled propane the temperature increase in the facility lines for slow and fast fill, assuming the same line is used for slow and fast fill, was assumed to be 4.2°k and 1.1°k, respectively. Subcooled propane was assumed to be at 91.5°k with a density of 16.48 mol/L, an enthalpy of -21352 J/mol, a heat of vaporization of 24540 J/mol, a viscosity of 2.399 poise, and a specific heat of 92.2 J/mol-°k. Normal boiling point propane was at 231.04 °k with a density of 13.18 mol/L, an enthalpy of -8793 J/mol, a viscosity of 2.399 poise, and a specific heat of 84.6 J/mol-°k. The engine interface point where autogenous pressurant is obtained had temperature, pressure and enthalpy for subcooled and NBP propane values of 230°k/31.5 MPa/-7479 J/mol and 340°k/22.7 MPa/3704 J/mol, respectively. Autogenous conditions were determined from Reference 1 for subcooled and NBP engines operating at 20.7 and 18.6 MPa chamber pressure and vacuum thrust of approximately 3114 KN. Absolute values for the autogenous pressurant enthalpies were obtained by determining the value for enthalpy at absolute zero for propane, which is -27230 J/mol. Therefore, the autogenous pressurant engine interface enthalpies with respect to absolute zero for subcooled and NBP

19751 and 30934 J/mol respectively. Propane properties at various states were obtained from the NBS (National Bureau of Standard) propellant properties program (MIPROPS).

4.3.3 Discussion of Analysis and Results

Table 4.3-1 shows the vehicle subsystem effects from using subcooled versus NBP propane. The specific effects shown on Table 4.3-1 will be expounded upon below.

4.3.3.1 Vehicle Subsystems

Instrumentation

Subcooled and NBP propane requires cryogenic and non-cryogenic instrumentation, respectively. Instrumentation effected consists of only temperature and level sensors, since cryogenic pressure instrumentation consists primarily of non-cryogenic probes with a pressure sense line.

Propellant Feedlines

Lower viscosity subcooled propane gives lower Reynolds Number flows for the same mass flowrate and line diameter, thereby giving a larger pressure drop. Therefore, a larger feedline would be used to reduce the pressure drop in the feed system.

Anti-Geyser System

Subcooled propane would require a larger heat input into the feedline to create a geyser effect; therefore, the anti-geyser system (which would probably be a helium bubbling system such as that used for the STS LO2 feedline) could be removed if the only purpose it served was to provide an anti-geyser effect. However, if a helium bubbling system was used to provide propellant conditioning (as shown on Table 4.3-2) then the system might remain and serve both purposes or be relocated to the bottom of the fuel tank and serve only as a propellant conditioning system.

Leak Detection

Leak detection systems required depend on the propellant conditioning techniques selected. If an ullage evacuation technique is used for propellant conditioning it would be necessary to investigate internal leak detection methods. Any other propellant conditioning technique would require similar leak detection equipment as on existing vehicles; however, under-pressure emergency safety procedures would most certainly be required to prevent internal leakage or tank structural failure.

Table 4.3-1 Vehicle Subsystems Impact

VEHICLE SUBSYSTEM	NBP PROPANE	SC PROPANE
INSTRUMENTATION (LOADING/FLIGHT)	NON-CRYOGENIC INSTRUMENTATION USED FOR TEMPERATURE AND LIQUID LEVEL INSTRUMENTATION	CRYOGENIC INSTRUMENTATION REQUIRED
PROPELLANT FEEDLINES	FEEDLINES SMALLER DUE TO LOWER VISCOSITY OF NBP PROPANE AND DO NOT REQUIRE CRYOGENIC CERTIFICATION	LARGER DIAMETER CRYOGENIC FEEDLINES REQUIRED
ANTI-GEYSER SYSTEM	ANTI-GEYSER SYSTEM SIMILAR TO THOSE ON EXISTING VEHICLES	ANTI-GEYSER SYSTEM MAY NOT BE REQUIRED DUE TO SUBCOOLED CONDITIONS
LEAK DETECTION	LEAK DETECTION SYSTEMS SIMILAR TO THOSE ON EXISTING VEHICLES	INTERNAL LEAK DETECTION METHODS OR UNDER-PRESSURE EMERGENCY PROCEDURES WOULD HAVE TO BE ESTABLISHED IF VACUUM CONDITIONS WERE NOT A BASELINED CAPABILITY FOR THE TANKAGE, FEED AND PRESSURIZATION SYSTEMS

Table 4.3-1 Vehicle Subsystems Impact (cont.)

VEHICLE SUBSYSTEM	NBP PROPANE	SC PROPANE
PURGE SYSTEMS	LESS HEATED PURGE FLOWRATE OR TEMPERATURE REQUIRED FOR COMPARTMENTS IN CONTACT WITH ONLY NBP PROPANE TANKAGE	GREATER HEATED PURGE REQUIREMENTS FOR COMPARTMENTS
TANKAGE	SEVENTY PERCENT LESS INSULATION THICKNESS REQUIRED AND TEN PERCENT GREATER TANKAGE SURFACE AREA RESULTS IN APPROXIMATELY SIXTY PERCENT LESS INSULATION BY VOLUME FOR THE NBP PROPANE TANKAGE VERSUS SUBCOOLED PROPANE	TWENTY PERCENT LESS VOLUME REQUIRED FOR SC PROPANE TANKAGE THAN FOR NBP PROPANE TANKS REQUIRED TO WITHSTAND MINOR OR AT MOST FULL VACUUM CONDITIONS
GROUND PRESSURIZATION SYSTEM	HE AND/OR N2 GROUND PRESS SYSTEMS SIMILAR TO EXISTING VEHICLES	N2 ABSORPTION INCREASES AT SUBCOOLED CONDITIONS IF VEHICLE NOT DESIGNED FOR VACUUM CONDITONS A GROUND PRESS SYSTEM FAILURE COULD BE CATASTROPHIC
FLIGHT PRESSURIZATION SYSTEM	GREATER PROBLEMS WITH REGENERATIVE CHANNEL COKING	FIFTY PERCENT GREATER AUTOGENOUS PRESSURANT REQUIREMENTS, IN TOTAL MASS, EXPECTED VERSUS NBP PROPANE

Purge System

Purge system requirements would be greater for subcooled propane in compartments which contact the propane tank only, such as a nose cone compartment area. Compartments which contact the LO2 tank or LO2 feedlines such as the intertank or engine compartments would have similar purge requirements for subcooled or NBP propane vehicles.

Tankage

Assuming equivalent heat inputs into the propane tankage, the insulation thickness was assumed to be proportional to the delta temperature across the insulation, giving a seventy percent reduction in insulation thickness requirements for the NBP propane versus subcooled propane tankage. However, NBP propane tankage has twenty five percent greater volume (subcooled propane twenty percent less volume than NBP tankage) and approximately ten percent greater surface area which results in approximately sixty percent less insulation by volume for the NBP propane tankage.

Subcooled propane tankage has one additional disadvantage due to its low vapor pressure. Low vapor pressure will drive the requirement for tankage designs that withstand minor vacuum conditions (to preclude a structural failure due to a small under-pressure condition during loading) or at most full vacuum conditions (for an evacuated ullage propellant conditioning technique).

Ground Pressurization System

Nitrogen absorption into the propane increases at subcooled conditions so helium would probably be used. But more importantly, the failure of pressurization lines during loading could possibly cause stage or vehicle destruction due to the possibility of under-pressure conditions and tank structural failure (dependent on tank structural designs).

Flight Pressurization System

To determine the effects on autogenous pressurant a rough look at the controlling factors will be investigated. The effects due to subcooled propane on tankage volume, engine autogenous interface conditions, and heat transfer from ullage to liquid surface are assumed to be as follows, all relative to tankage for NBP propane: 1) 20% less tankage volume is required, 2) 36% less engine autogenous interface enthalpies and 3) delta temperatures between incoming autogenous pressurant and the liquid surface are assumed to be 140°k and 110°k for subcooled and NBP propane. This gives a 27% greater temperature differential and, at most, a 15% greater heat transfer rate to the liquid surface due to ullage stratification. Using the estimates above a 50% increase in autogenous pressurant mass requirements would occur for subcooled versus NBP propane vehicles (AUTO REQ = .80 / .64 / .85 ≈ 1.50).

4.3.3.2 Propellant Conditioning Comparison

Table 4.3-2 shows the propellant conditioning techniques evaluated for the subcooled propane vehicle; NBP propane vehicles would use an ullage evacuation technique. If ullage evacuation is used for subcooled propane then tankage must be designed to withstand full vacuum conditions (.101 MPa) and interior leak detection methods would require further investigation. Although other conditioning techniques do not require the provisions mentioned for an ullage evacuation technique, these techniques may require vacuum condition safety procedures. Additional vehicle impacts discussed on Table 4.3-2 will be expounded upon below.

No Conditioning

The allowable heat leak for the STS LO2 tank is 665000 Btu/Hr or 11.7 MJ/min, as noted above. Based on the specific heat of subcooled propane of 84.6 J/mol-°k (1.92 J/g-°k) and a tank propellant load of 150167 kg, a temperature rise of 1°k would occur every 25 minutes (ΔTemp/Time = 11700000 / 1920 / 150167 = 0.04058 °k/min).

Ullage Evacuation

Using a total vehicle subcooled propellant load of 300334 kg. Based on the STS heat load of 11.7 MJ/min (total subcooled propellant heat load of 23.4 MJ/min) and the heat of vaporization of 24540 J/mol or 556.6 J/g the vehicle evacuation rate would be 42 kg/min (mdot/time = 23400000 / 556600 = 42.0 kg/min).

Recirculated Flow

For 100% mixing efficiency, density of 16.48 mol/L, specific heat of 92.2 J/mol-°k, and total vehicle heat load of 23.4 MJ/min a temperature differential of 12.3°k and 2.0°k between storage tank and vehicle propellant can be maintained for recirculation flowrates of 1900 and 17000 L/min (slow fill ΔTemp = 23400000 / 1900 / 92.2 / 16.48 + 4.2(facility ΔT) = 12.3°k, fast fill ΔTemp = 23400000/ 17000 / 92.2 / 16.48 + 1.1(facility ΔT) = 2.0°k). Internal tank circulation techniques would require further investigation.

Helium Bubbling

STS LO2 tank experience has shown that helium bubbling used for anti-geyser protection also provides some bulk propellant conditioning. This same technique can be used to condition subcooled propane with higher expected flowrates than that required for the STS system. The feasibility of helium bubbling conditioning for large quantities of subcooled propane would have to be investigated further using analysis and experimentation. Furthermore, if helium bubbling conditioning is possible with large helium flowrates a closed loop purification system could be investigated to limit helium requirements.

Liquid Surface Forced Convection

Liquid evaporation and thereby liquid surface cooling and possibly bulk propellant conditioning can be induced by creating forced convection with nitrogen or helium on the liquid surface. Although the technique seems plausible the flowrates required would be quite high (driving the requirement for a closed loop purification system) and the ability to condition large quantities of propellant is questionable.

4.3.4 Conclusions

Most flight subsystems are not severely impacted no matter what the conditioning technique. Cost differences between NBP propane and subcooled propane in these areas will be minor. However, several flight subsystems will experience major impacts, when comparing subcooled and NBP propane vehicles, dependent upon the selected propellant conditioning techniques.

An ullage evacuation approach would lead to excessive vehicle impacts and complexity. Vacuum condition (vehicle under-pressure) safety procedures and some other propellant conditioning technique should be investigated. The best approach would be to use a helium bubbling technique for bulk propellant conditioning. Helium bubbling can maintain the desired propellant condition and provide launch hold capabilities with less complexity than any other technique. The only disadvantage with a helium bubbling system is the possibility of high helium usage requirements which could require the development of a closed loop purification system.

However, it is expected that whatever propellant conditioning technique is used, besides ullage evacuation, the impact on the vehicle mass should be less than one percent of the total dry mass. Cost impacts would be negligible when compared to overall vehicle cost.

Table 4.3-2 Propellant Conditioning Impacts

PROPELLANT CONDITIONING	VEHICLE IMPACTS
NO CONDITIONING	1^{0}K TEMPERATURE RISE EVERY 25 MINUTES FOR A TANK HEAT LOAD OF 11.7 MJ/MIN (184.7 BTU/SEC - ET LO2 TANK ALLOWABLE HEAT LEAK) VACUUM CONDITION SAFING PROCEDURES REQUIRED
ULLAGE EVACUATION	EVACUATION RATE OF 42 KG/MIN REQUIRED FOR A VEHICLE HEAT LOAD OF 23.4 MJ/MIN TANKS DESIGNED TO WITHSTAND UP TO .101 MPa VACUUM CONDITION PROVISIONS FOR INTERIOR LEAK DETECTION REQUIRE INVESTIGATION
RECIRCULATED FLOW	RECIRCULATION FLOWRATES OF 1900 AND 17000 L/MIN WOULD MAINTAIN A TEMPERATURE DIFFERENTIAL OF 12.3 ^{0}K AND 2.0 ^{0}K BETWEEN STORAGE TANK AND VEHICLE PROPELLANT FOR A VEHICLE HEAT LOAD OF 23.4 MJ/MIN AND 100% MIXING EFFICIENCY RECIRCULATION INTERFACE TO FACILITY REQUIRED PROPELLANT MIXING TECHNIQUES REQUIRE INVESTIGATION VACUUM CONDITION SAFING PROCEDURES REQUIRED

Table 4.3-2 Propellant Conditioning Impacts (cont.)

PROPELLANT CONDITIONING	VEHICLE IMPACTS
HELIUM BUBBLING	REQUIRED HELIUM FLOWRATES WOULD HAVE TO BE INVESTIGATED HIGH HELIUM FLOWRATES MAY CONSTITUTE THE USE OF A CLOSED LOOP PURIFICATION SYSTEM VACUUM CONDITION SAFING PROCEDURES REQUIRED
LIQUID SURFACE FORCED CONVECTION	FORCED CONVECTION TECHNIQUES REQUIRE INVESTIGATION HIGH FLOWRATES (HE OR N2) WOULD MOST LIKELY BE REQUIRED, DRIVING THE REQUIREMENT FOR A CLOSED LOOP PURIFICATION SYSTEM SUCCESS OF FORCED CONVECTION TECHNIQUE QUESTIONABLE VACUUM CONDITION SAFING PROCEDURES REQUIRED

5.0 CONCLUSIONS AND RECOMMENDATIONS

5.1 Task 1

On the basis of the Trades analysis, the best fuel/coolant option for the UFRCV is subcooled propane, or methane, with fuel cooling. The best option for the SSTO is subcooled propane with hydrogen cooling although the reduction in total vehicle dry mass from the reference vehicle is only seven percent. We recommend subcooled propane as the best fuel for any coolant combination; although methane is almost as good a performer and lacks some of the negative aspects of subcooled propane when ground support systems are examined, see below.

We recommend the use of hydrogen cooled engines only for the SSTO and when cross feeding propellants from the booster to the second stage of a two stage vehicle. For the latter, the best fuel option is subcooled propane.

Although the combined use of cross feeding propellants and hydrogen cooling generated the greatest mass reductions for the UFRCV, the result was only marginally better than for the recommended fuel/coolant options above. This marginal improvement may be outweighed by the added complexity due to the presence of three propellants on the stage and the cross feeding of propellants. The overall impact on the whole system, including ground operations, may be more significant than the marginal mass reduction improvement in terms of other evaluation criteria such as cost. However, on strictly a total vehicle dry mass basis, we can recommend that if hydrogen cooling is used in the boost stage, then cross feeding propellants should be used.

From our analysis, we cannot recommend the use of either high mixture ratio or variable mixture ratio LOX/LH2 engines for use in the boost stage, or boost phase, of a launch vehicle.

From the results of the translating nozzle analysis, it is apparent that for the ground rules we utilized, the use of a translating nozzle for a boost engine is not appropriate. It may be possible to find different values for expansion ratio that would make this option marginally better than not having a translating nozzle. However, due to the small fraction of the boost phase compared to the entire vehicle flight time, we doubt that any combination of expansion ratio values exist that significantly improve upon a single expansion ratio value selected to minimize vehicle dry mass impact.

From our sensitivity analysis, we can specify some ranges of engine parameters that do not affect the vehicle materially in dry mass. On a total vehicle dry mass basis then, any change of engine parameters in these ranges is unimportant to the vehicle design and do not have to be evaluated as having an impact on vehicle design. This allows a measure of uncoupling of vehicle and engine design. However, this presumes that the vehicle design is created with an awareness of these parameter ranges. If the vehicle design is generated with a specific engine performance level in mind, any shift in engine parameters from this specific level could have a disastrous impact on vehicle dry mass. The specific ranges, all of which generate changes in vehicle total dry mass of less than ±two percent, are: ±two percent vacuum specific impulse, ±seven percent on engine thrust to weight ratio (installed), and ±thirty percent of engine mixture ratio (hydrocarbon fuels).

Overall, the best mass reductions from the all hydrogen vehicle were in the vicinity of fourteen percent, with most reductions below ten percent. Previous studies considered a vehicle dry mass reduction of approximately twenty percent or more as significant when cost was the criteria for evaluation.[9] It remains to be seen whether the analysis presented here can adequately justify the fuel/coolant combination to be used for the two types of vehicles examined when the whole transportation system is the issue.

The above recommendations are valid for the two types of vehicles examined in this study and for boost phase propulsion. It may be possible to obtain greater mass reductions, from an all LOX/LH2 vehicle, by using the various fuel/coolant options in the second stage, or sustainer phase, as well as in the boost phase. This is especially promising because added possibilities exist for cross feeding propellants in a two stage vehicle. It is recommended that a similar study by done to examine these possibilities.

It is unlikely that the results presented here can be extended to other types of vehicles. Particularly for expendable and for partially reusable two stage vehicles, the design differences from the vehicles examined in this study are significant. It is also recommended that a study, similar to this one, be conducted to examine the impact on these other types of launch vehicles.

5.2 Task 2.

The most significant impacts on ground support systems is in the area of storing the subcooled propane and its refrigeration, both of which substantially increase the cost relative to the use of NBP propane. However, we used a worst case method in our analysis by assuming that the propane would be subcooled the entire time from just after delivery to transfer to the vehicle. It may be readily possible to mitigate some of the cost impacts by selecting alternate methods of storage etc.

With the largest difference between the two fuel types being only five million dollars, we do not believe that the impact on ground support systems due to using subcooled propane versus NBP propane does not allow significant discrimination between the two propane states. The dollar value is insignificant when compared to the sizable cost of building new or modifying existing launch facilities and is smaller still when compared to the cost of obtaining a new transportation system. Attempting to use this issue to preferentially select one type of propane over another is not appropriate.

We consider that the impacts on vehicle flight subsystems due to the use of subcooled propane, as compared to NBP propane, to be minor for the most part.

We estimated the overall vehicle dry mass impact to be less than one percent of the total, with a comparable value for the cost impact, and this value is negligible when compared to other issues. As in the case of the ground support systems, further investigation of the issue of vehicle impact, in this area, as a discrimination to allow a "best" selection of hydrocarbon fuel should not be pursued.

We do recommend that additional research be conducted to define the optimal choice of propellant conditioning on the vehicle, both on the ground and during flight. In our analysis, the best alternative for propellant conditioning was not clear. An integrated systems approach is necessary to properly evaluate competing alternatives on the basis of performance, safety and cost. However, we do not believe this issue to be unsolvable on a technical basis and the cost impact on the vehicle is expected to be minor relative to other systems.

APPENDIX A

Detailed Mass Data for Optimum Configurations

This appendix contains the detailed mass data for the optimum configurations determined during the trades analysis, see Section 3.4.4.2. Also provided is the detailed mass data for the reference, all LOX/LH2, vehicles, see Section 3.4.4.1. The data consists of the key sheets from the respective sizing program's output. The units of mass are shown on the sheets for the SSTO configurations. The units for all the UFRCV outputs are English units, specifically pounds for all weights shown. Other units are noted in the output. The following table of contents is provided to locate the appropriate mass data for a specific vehicle.

H/H SSTO

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                    *************************
                    * M A S S   R E P O R T *
                    *************************
       CASE  81    TRAJECTORY BURNOUT MASS=    144524.20  kg
     BOEING SINGLE-STAGE-TO-ORBIT CONCEPT  -- 13.6 METRIC TON PAYLOAD
 1.0 WING GROUP                                           8341.  kg
 2.0 TAIL GROUP                                           1802.  kg
 3.0 BODY GROUP                                          28474.  kg
      BASIC STRUCTURE                       9351.  kg
      THRUST STRUCTURE                      3440.  kg
      RP-1 TANK                                0.  kg
      LOX TANK                              6490.  kg
      LH2 TANK                              8565.  kg
      BODY FLAP                              629.  kg
 4.0 INDUCED ENVIRONMENT                                 13138.  kg
 5.0 LANDING GEAR                                         3931.  kg
 6.0 PROPULSION                                          28820.  kg
 7.0 PROPULSION, RCS                                      1312.  kg
 8.0 PROPULSION, OMS                                      1455.  kg
 9.0 PRIME POWER                                          1428.  kg
10.0 ELEC CONV AND DISTR                                  1957.  kg
11.0 HYDRAULICS AND SURFACE CONTROLS                      6019.  kg
13.0 AVIONICS                                             2248.  kg
14.0 ENVIRONMENTAL CONTROL                                1989.  kg
15.0 PERSONNEL PROVISIONS                                  763.  kg
16.0 MARGIN                                               7286.  kg
    DRY WEIGHT                                          108962.  kg ( .765 1)

17.0 PERSONNEL                                            1290.  kg
19.0 RESIDUAL FLUIDS                                      5819.  kg
    LANDED WEIGHT W/O CARGO                             116071.  kg ( .766 1)

20.0 CARGO (RETURNED)                                    13600.  kg
    LANDED WEIGHT                                       129671.  kg ( .747 1)
    ENTRY WEIGHT                                        129671.  kg ( .747 1)
23.0 ACPS PROPELLANT                                     11333.  kg
              RCS                2684.  kg
              OMS                8649.  kg
24.0 CARGO DELIVERED                                         0.  kg
25.0 ASCENT RESERVES                                      2686.  kg
26.0 INFLIGHT LOSSES                                       868.  kg
27.0 ASCENT PROPELLANT                                  895216.  kg
      HC             0.  kg LOX ENG B          0.  kg  HC %=  .0
      LH2        99467.  kg LOX ENG A     795748.  kg
      HC ENGINES        .0    H2 ENGINES       6.7
      HC THRUST PER ENGINE kN 2224.1
      H2 THRUST PER ENGINE kN 2224.1

    GROSS LIFT OFF MASS                                1039774.  kg ( .103 1)


                 ***************
                 * DESIGN DATA *
                 ***************

STRUCTURAL MASS REDUCTION          .25
BODY LENGTH                      48.67 m
VERTICAL TAIL AREA               74.51 sqm
THEORETICAL WING AREA           389.46 sqm
WING SPAN                        32.12 m
STRUCTURAL SPAN                  19.07 m
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R/R SSTO

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                    *************************
                    * M A S S   R E P O R T *
                    *************************
     CASE 273    TRAJECTORY BURNOUT MASS=    155736.40  kg
   BOEING SINGLE-STAGE-TO-ORBIT CONCEPT  -- 13.6 METRIC TON PAYLOAD
 1.0 WING GROUP                                        9070.  kg
 2.0 TAIL GROUP                                        1913.  kg
 3.0 BODY GROUP                                       32474.  kg
      BASIC STRUCTURE                     9915.  kg
      THRUST STRUCTURE                    4050.  kg
      RP-1 TANK                           2120.  kg
      LOX TANK                            6987.  kg
      LH2 TANK                            8736.  kg
      BODY FLAP                            665.  kg
 4.0 INDUCED ENVIRONMENT                              13904.  kg
 5.0 LANDING GEAR                                      4220.  kg
 6.0 PROPULSION                                       30188.  kg
 7.0 PROPULSION, RCS                                   1409.  kg
 8.0 PROPULSION, OMS                                   1563.  kg
 9.0 PRIME POWER                                       1428.  kg
10.0 ELEC CONV AND DISTR                               1962.  kg
11.0 HYDRAULICS AND SURFACE CONTROLS                   6349.  kg
13.0 AVIONICS                                          2248.  kg
14.0 ENVIRONMENTAL CONTROL                             1989.  kg
15.0 PERSONNEL PROVISIONS                               763.  kg
16.0 MARGIN                                            7929.  kg
    DRY WEIGHT                                       117409.  kg ( .764 1)

17.0 PERSONNEL                                         1290.  kg
19.0 RESIDUAL FLUIDS                                   6967.  kg
    LANDED WEIGHT W/O CARGO                          125666.  kg ( .766 1)

20.0 CARGO (RETURNED)                                 13600.  kg
    LANDED WEIGHT                                    139266.  kg ( .749 1)
    ENTRY WEIGHT                                     139266.  kg ( .749 1)
23.0 ACPS PROPELLANT                                  12172.  kg
              RCS                 2883.  kg
              OMS                 9289.  kg
24.0 CARGO DELIVERED                                      0.  kg
25.0 ASCENT RESERVES                                   3216.  kg
26.0 INFLIGHT LOSSES                                   1040.  kg
27.0 ASCENT PROPELLANT                              1071868.  kg
      HC          105565.  kg LOX ENG B      255465.  kg  HC %=33.7
      LH2         101550.  kg LOX ENG A      609288.  kg
      HC ENGINES        2.8    H2 ENGINES      4.0
      HC THRUST PER ENGINE kN 3199.0
      H2 THRUST PER ENGINE kN 2224.1

    GROSS LIFT OFF MASS                             1227561.  kg ( .094 1)


                 ***************
                 * DESIGN DATA *
                 ***************

STRUCTURAL MASS REDUCTION          .25
SUBSYSTEM MASS REDUCTION           .15
BODY LENGTH                      50.07 m
VERTICAL TAIL AREA               78.85 sqm
THEORETICAL WING AREA           412.17 sqm
WING SPAN                        33.05 m
STRUCTURAL SPAN                  19.62 m
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ORIGINAL PAGE IS OF POOR QUALITY

M/M SSTO

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* M A S S   R E P O R T *
****

CASE 368   TRAJECTORY BURNOUT MASS: 146035.00 kg

BOEING SINGLE-STAGE-TO-ORBIT CONCEPT -- 13.6 METRIC TON PAYLOAD

1.0 WING GROUP		8287. kg
2.0 TAIL GROUP		1773. kg
3.0 BODY GROUP		31277. kg
BASIC STRUCTURE	9205. kg	
THRUST STRUCTURE	3916. kg	
RP-1 TANK	4668. kg	
LOX TANK	6910. kg	
LH2 TANK	5959. kg	
BODY FLAP	619. kg	
4.0 INDUCED ENVIRONMENT		12939. kg
5.0 LANDING GEAR		3951. kg
6.0 PROPULSION		25662. kg
7.0 PROPULSION, RCS		1319. kg
8.0 PROPULSION, OMS		1463. kg
9.0 PRIME POWER		1428. kg
10.0 ELEC CONV AND DISTR		1956. kg
11.0 HYDRAULICS AND SURFACE CONTROLS		5933. kg
13.0 AVIONICS		2248. kg
14.0 ENVIRONMENTAL CONTROL		1389. kg
15.0 PERSONNEL PROVISIONS		763. kg
16.0 MARGIN		7533. kg
DRY WEIGHT		108521. kg (.756 1)
17.0 PERSONNEL		1290. kg
19.0 RESIDUAL FLUIDS		6942. kg
LANDED WEIGHT W/O CARGO		116753. kg (.759 1)
20.0 CARGO (RETURNED)		13600. kg
LANDED WEIGHT		130353. kg (.741 1)
ENTRY WEIGHT		130353. kg (.741 1)
23.0 ACPS PROPELLANT		11393. kg
RCS	2698. kg	
OMS	8695. kg	
24.0 CARGO DELIVERED		0. kg
25.0 ASCENT RESERVES		3204. kg
26.0 INFLIGHT LOSSES		1036. kg
27.0 ASCENT PROPELLANT		1068010. kg

HC	145770. kg	LOX ENG B	444630. kg	HC %=55.3
LH2	68231. kg	LOX ENG A	409379. kg	
HC ENGINES	4.0	H2 ENGINES	2.4	

HC THRUST PER ENGINE kN 3108.0
H2 THRUST PER ENGINE kN 2224.1

GROSS LIFT OFF MASS   1213996. kg (.088 1)

****
* DESIGN DATA *
****

STRUCTURAL MASS REDUCTION	.25
BODY LENGTH	48.30 m
VERTICAL TAIL AREA	73.38 sqm
THEORETICAL WING AREA	383.57 sqm
WING SPAN	31.88 m
STRUCTURAL SPAN	18.93 m

NP/NP SSTO

MASS REPORT

CASE 494 TRAJECTORY BURNOUT MASS= 151080.70 kg

BOEING SINGLE-STAGE-TO-ORBIT CONCEPT -- 13.6 METRIC TON PAYLOAD

1.0 WING GROUP		8730. kg	
2.0 TAIL GROUP		1859. kg	
3.0 BODY GROUP		32034. kg	
BASIC STRUCTURE	9640. kg		
THRUST STRUCTURE	3965. kg		
RP-1 TANK	3196. kg		
LOX TANK	6927. kg		
LH2 TANK	7658. kg		
BODY FLAP	647. kg		
4.0 INDUCED ENVIRONMENT		13530. kg	
5.0 LANDING GEAR		4091. kg	
6.0 PROPULSION		27676. kg	
7.0 PROPULSION, RCS		1365. kg	
8.0 PROPULSION, OMS		1515. kg	
9.0 PRIME POWER		1428. kg	
10.0 ELEC CONV AND DISTR		1959. kg	
11.0 HYDRAULICS AND SURFACE CONTROLS		6188. kg	
13.0 AVIONICS		2248. kg	
14.0 ENVIRONMENTAL CONTROL		1939. kg	
15.0 PERSONNEL PROVISIONS		763. kg	
16.0 MARGIN		7770. kg	
DRY WEIGHT		113145. kg	(.759 1)
17.0 PERSONNEL		1290. kg	
19.0 RESIDUAL FLUIDS		6949. kg	
LANDED WEIGHT W/O CARGO		121384. kg	(.762 1)
20.0 CARGO (RETURNED)		13600. kg	
LANDED WEIGHT		134984. kg	(.744 1)
ENTRY WEIGHT		134984. kg	(.744 1)
23.0 ACPS PROPELLANT		11798. kg	
RCS	2794. kg		
OMS	9003. kg		
24.0 CARGO DELIVERED		0. kg	
25.0 ASCENT RESERVES		3207. kg	
26.0 INFLIGHT LOSSES		1037. kg	
27.0 ASCENT PROPELLANT		1069084. kg	

HC	124068. kg	LOX ENG B	325127. kg	HC %=42.0
LH2	88557. kg	LOX ENG A	531331. kg	
HC ENGINES	3.4	H2 ENGINES	3.2	

HC THRUST PER ENGINE kN 3148.0

H2 THRUST PER ENGINE kN 2224.1

GROSS LIFT OFF MASS 1220110. kg (.091 1)

*** DESIGN DATA ***

STRUCTURAL MASS REDUCTION	.25
BODY LENGTH	49.39 m
VERTICAL TAIL AREA	76.74 sqm
THEORETICAL WING AREA	401.09 sqm
WING SPAN	32.60 m
STRUCTURAL SPAN	19.35 m

SP/SP SSTO

****
MASS REPORT
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CASE 568 TRAJECTORY BURNOUT MASS= 140504.20 kg
BOEING SINGLE-STAGE-TO-ORBIT CONCEPT -- 13.6 METRIC TON PAYLOAD

1.0 WING GROUP		7870. kg
2.0 TAIL GROUP		1700. kg
3.0 BODY GROUP		29254. kg
BASIC STRUCTURE	8833. kg	
THRUST STRUCTURE	3890. kg	
RP-1 TANK	3273. kg	
LOX TANK	6784. kg	
LH2 TANK	5880. kg	
BODY FLAP	595. kg	
4.0 INDUCED ENVIRONMENT		12432. kg
5.0 LANDING GEAR		3801. kg
6.0 PROPULSION		24561. kg
7.0 PROPULSION, RCS		1268. kg
8.0 PROPULSION, OMS		1407. kg
9.0 PRIME POWER		1428. kg
10.0 ELEC CONV AND DISTR		1952. kg
11.0 HYDRAULICS AND SURFACE CONTROLS		5713. kg
13.0 AVIONICS		2248. kg
14.0 ENVIRONMENTAL CONTROL		1989. kg
15.0 PERSONNEL PROVISIONS		763. kg
16.0 MARGIN		7183. kg
DRY WEIGHT		103569. kg (.756 1)
17.0 PERSONNEL		1290. kg
19.0 RESIDUAL FLUIDS		6916. kg
LANDED WEIGHT W/O CARGO		111775. kg (.759 1)
20.0 CARGO (RETURNED)		13600. kg
LANDED WEIGHT		125375. kg (.741 1)
ENTRY WEIGHT		125375. kg (.741 1)
23.0 ACPS PROPELLANT		10958. kg
RCS	2595. kg	
OMS	8363. kg	
24.0 CARGO DELIVERED		0. kg
25.0 ASCENT RESERVES		3192. kg
26.0 INFLIGHT LOSSES		1032. kg
27.0 ASCENT PROPELLANT		1064049. kg

HC 160320. kg LOX ENG B 432798. kg HC %=55.7
LH2 67277. kg LOX ENG A 403655. kg
HC ENGINES 3.9 H2 ENGINES 2.4
HC THRUST PER ENGINE kN 3111.0
H2 THRUST PER ENGINE kN 2224.1

GROSS LIFT OFF MASS 1204606. kg (.085 1)

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* DESIGN DATA *
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STRUCTURAL MASS REDUCTION	.25
BODY LENGTH	47.35 m
VERTICAL TAIL AREA	70.50 sqm
THEORETICAL WING AREA	368.53 sqm
WING SPAN	31.25 m
STRUCTURAL SPAN	18.55 m

R/H SSTO

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                    * M A S S   R E P O R T *
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     CASE 638    TRAJECTORY BURNOUT MASS=    141373.10  kg
   BOEING SINGLE-STAGE-TO-ORBIT CONCEPT  -- 13.6 METRIC TON PAYLOAD
 1.0 WING GROUP                                        7993.  kg
 2.0 TAIL GROUP                                        1732.  kg
 3.0 BODY GROUP                                       29601.  kg
      BASIC STRUCTURE                    8994.  kg
      THRUST STRUCTURE                   3881.  kg
      RP-1 TANK                          2933.  kg
      LOX TANK                           6790.  kg
      LH2 TANK                           6398.  kg
      BODY FLAP                           605.  kg
 4.0 INDUCED ENVIRONMENT                              12652.  kg
 5.0 LANDING GEAR                                      3825.  kg
 6.0 PROPULSION                                       24395.  kg
 7.0 PROPULSION, RCS                                   1276.  kg
 8.0 PROPULSION, OMS                                   1416.  kg
 9.0 PRIME POWER                                       1428.  kg
10.0 ELEC CONV AND DISTR                               1954.  kg
11.0 HYDRAULICS AND SURFACE CONTROLS                   5808.  kg
13.0 AVIONICS                                          2248.  kg
14.0 ENVIRONMENTAL CONTROL                             1989.  kg
15.0 PERSONNEL PROVISIONS                               763.  kg
16.0 MARGIN                                            7269.  kg
   DRY WEIGHT                                        104349.  kg ( .755 1)

17.0 PERSONNEL                                         1290.  kg
19.0 RESIDUAL FLUIDS                                   6924.  kg
   LANDED WEIGHT W/O CARGO                           112563.  kg ( .758 1)

20.0 CARGO (RETURNED)                                 13600.  kg
   LANDED WEIGHT                                     126163.  kg ( .740 1)
   ENTRY WEIGHT                                      126163.  kg ( .740 1)
23.0 ACPS PROPELLANT                                  11027.  kg
            RCS              2612.  kg
            OMS              8415.  kg
24.0 CARGO DELIVERED                                      0.  kg
25.0 ASCENT RESERVES                                   3196.  kg
26.0 INFLIGHT LOSSES                                   1033.  kg
27.0 ASCENT PROPELLANT                              1065221.  kg
      HC        154476.  kg LOX ENG B     432534.  kg  HC %=55.1
      LH2        73451.  kg LOX ENG A     404760.  kg
      HC ENGINES       4.0    H2 ENGINES     2.4
      HC THRUST PER ENGINE kN 3090.0
      H2 THRUST PER ENGINE kN 2224.1

    GROSS LIFT OFF MASS                             1206639.  kg ( .086 1)


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               * DESIGN DATA *
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STRUCTURAL MASS REDUCTION        .25
BODY LENGTH                    47.76 m
VERTICAL TAIL AREA             71.75 sqm
THEORETICAL WING AREA         375.04 sqm
WING SPAN                      31.52 m
STRUCTURAL SPAN                18.71 m
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M A S S   R E P O R T
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CASE 749   TRAJECTORY BURNOUT MASS=   142971.20 kg

BOEING SINGLE-STAGE-TO-ORBIT CONCEPT -- 13.6 METRIC TON PAYLOAD

1.0 WING GROUP		8095. kg	
2.0 TAIL GROUP		1745. kg	
3.0 BODY GROUP		30770. kg	
BASIC STRUCTURE	9063. kg		
THRUST STRUCTURE	3877. kg		
RP-1 TANK	4686. kg		
LOX TANK	6912. kg		
LH2 TANK	5622. kg		
BODY FLAP	610. kg		
4.0 INDUCED ENVIRONMENT		12745. kg	
5.0 LANDING GEAR		3871. kg	
6.0 PROPULSION		24282. kg	
7.0 PROPULSION, RCS		1292. kg	
8.0 PROPULSION, OMS		1433. kg	
9.0 PRIME POWER		1428. kg	
10.0 ELEC CONV AND DISTR		1954. kg	
11.0 HYDRAULICS AND SURFACE CONTROLS		5849. kg	
13.0 AVIONICS		2248. kg	
14.0 ENVIRONMENTAL CONTROL		1989. kg	
15.0 PERSONNEL PROVISIONS		763. kg	
16.0 MARGIN		7418. kg	
DRY WEIGHT		105883. kg	(.753 1)
17.0 PERSONNEL		1290. kg	
19.0 RESIDUAL FLUIDS		6921. kg	
LANDED WEIGHT W/O CARGO		114094. kg	(.756 1)
20.0 CARGO (RETURNED)		13600. kg	
LANDED WEIGHT		127694. kg	(.739 1)
ENTRY WEIGHT		127694. kg	(.739 1)
23.0 ACPS PROPELLANT		11160. kg	
RCS	2643. kg		
OMS	8517. kg		
24.0 CARGO DELIVERED		0. kg	
25.0 ASCENT RESERVES		3194. kg	
26.0 INFLIGHT LOSSES		1033. kg	
27.0 ASCENT PROPELLANT		1064806. kg	

HC   146419. kg LOX ENG B   512467. kg HC %=61.9

LH2   64213. kg LOX ENG A   341706. kg

HC ENGINES   4.2   H2 ENGINES   2.0

HC THRUST PER ENGINE kN 3087.0

H2 THRUST PER ENGINE kN 2224.1

GROSS LIFT OFF MASS   1207888. kg (.087 1)

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* DESIGN DATA *
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STRUCTURAL MASS REDUCTION	.25
BODY LENGTH	47.94 m
VERTICAL TAIL AREA	72.28 sqm
THEORETICAL WING AREA	377.82 sqm
WING SPAN	31.64 m
STRUCTURAL SPAN	18.78 m

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                    * M A S S   R E P O R T *
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       CASE 838    TRAJECTORY BURNOUT MASS=    143830.00  kg
      BOEING SINGLE-STAGE-TO-ORBIT CONCEPT  -- 13.6 METRIC TON PAYLOAD
  1.0 WING GROUP                                          8174.  kg
  2.0 TAIL GROUP                                          1764.  kg
  3.0 BODY GROUP                                         30522.  kg
       BASIC STRUCTURE                       9159.  kg
       THRUST STRUCTURE                      3884.  kg
       RP-1 TANK                             3587.  kg
       LOX TANK                              6853.  kg
       LH2 TANK                              6422.  kg
       BODY FLAP                              616.  kg
  4.0 INDUCED ENVIRONMENT                                12877.  kg
  5.0 LANDING GEAR                                        3889.  kg
  6.0 PROPULSION                                         24803.  kg
  7.0 PROPULSION, RCS                                     1297.  kg
  8.0 PROPULSION, OMS                                     1439.  kg
  9.0 PRIME POWER                                         1428.  kg
 10.0 ELEC CONV AND DISTR                                 1955.  kg
 11.0 HYDRAULICS AND SURFACE CONTROLS                     5906.  kg
 13.0 AVIONICS                                            2248.  kg
 14.0 ENVIRONMENTAL CONTROL                               1989.  kg
 15.0 PERSONNEL PROVISIONS                                 763.  kg
 16.0 MARGIN                                              7425.  kg
    DRY WEIGHT                                          106480.  kg ( .755 l)

 17.0 PERSONNEL                                           1290.  kg
 19.0 RESIDUAL FLUIDS                                     6903.  kg
    LANDED WEIGHT W/O CARGO                             114673.  kg ( .758 l)

 20.0 CARGO (RETURNED)                                   13600.  kg
    LANDED WEIGHT                                       128273.  kg ( .740 l)
    ENTRY WEIGHT                                        128273.  kg ( .740 l)
 23.0 ACPS PROPELLANT                                    11211.  kg
                RCS                 2655.  kg
                OMS                 8556.  kg
 24.0 CARGO DELIVERED                                        0.  kg
 25.0 ASCENT RESERVES                                     3186.  kg
 26.0 INFLIGHT LOSSES                                     1030.  kg
 27.0 ASCENT PROPELLANT                                1062014.  kg
       HC          142082.  kg LOX ENG B      440373.  kg  HC %=54.8
       LH2          73757.  kg LOX ENG A      405802.  kg
       HC ENGINES        3.9    H2 ENGINES     2.4
       HC THRUST PER ENGINE kN 3095.0
       H2 THRUST PER ENGINE kN 2224.1

     GROSS LIFT OFF MASS                               1205714.  kg ( .087 l)


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               * DESIGN DATA *
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STRUCTURAL MASS REDUCTION          .25
BODY LENGTH                      48.19 m
VERTICAL TAIL AREA               73.03 sqm
THEORETICAL WING AREA           381.72 sqm
WING SPAN                        31.80 m
STRUCTURAL SPAN                  18.88 m
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MASS REPORT

CASE 939 TRAJECTORY BURNOUT MASS= 138589.10 kg

BOEING SINGLE-STAGE-TO-ORBIT CONCEPT -- 13.6 METRIC TON PAYLOAD

1.0 WING GROUP		7807. kg
2.0 TAIL GROUP		1703. kg
3.0 BODY GROUP		29129. kg
BASIC STRUCTURE	8846. kg	
THRUST STRUCTURE	3830. kg	
RP-1 TANK	3070. kg	
LOX TANK	6777. kg	
LH2 TANK	6010. kg	
BODY FLAP	596. kg	
4.0 INDUCED ENVIRONMENT		12451. kg
5.0 LANDING GEAR		3750. kg
6.0 PROPULSION		23195. kg
7.0 PROPULSION, RCS		1251. kg
8.0 PROPULSION, OMS		1388. kg
9.0 PRIME POWER		1428. kg
10.0 ELEC CONV AND DISTR		1952. kg
11.0 HYDRAULICS AND SURFACE CONTROLS		5721. kg
13.0 AVIONICS		2248. kg
14.0 ENVIRONMENTAL CONTROL		1989 kg
15.0 PERSONNEL PROVISIONS		763. kg
16.0 MARGIN		7158. kg
DRY WEIGHT		101934. kg (.753 1)
17.0 PERSONNEL		1290. kg
19.0 RESIDUAL FLUIDS		6845. kg
LANDED WEIGHT W/O CARGO		110068. kg (.756 1)
20.0 CARGO (RETURNED)		13600. kg
LANDED WEIGHT		123668. kg (.738 1)
ENTRY WEIGHT		123668. kg (.738 1)
23.0 ACPS PROPELLANT		10809. kg
RCS	2560. kg	
OMS	8249. kg	
24.0 CARGO DELIVERED		0. kg
25.0 ASCENT RESERVES		3159. kg
26.0 INFLIGHT LOSSES		1021. kg
27.0 ASCENT PROPELLANT		1053033. kg

HC	148719. kg	LOX ENG B	460942. kg	HC %=57.9
LH2	68832. kg	LOX ENG A	374540. kg	
HC ENGINES	4.2	H2 ENGINES	1.9	

HC THRUST PER ENGINE kN 3094.0

H2 THRUST PER ENGINE kN 2224.1

GROSS LIFT OFF MASS 1191691. kg (.085 1)

*** DESIGN DATA ***

STRUCTURAL MASS REDUCTION	.25
BODY LENGTH	47.38 m
VERTICAL TAIL AREA	70.61 sqm
THEORETICAL WING AREA	369.08 sqm
WING SPAN	31.27 m
STRUCTURAL SPAN	18.56 m

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PERFORMANCE PARAMETERS

NUMBER OF ITERATIONS 34

PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3367520.
THEORETICAL VELOCITY (FPS)	30147.
ACTUAL VELOCITY (FPS)	24551.
VELOCITY LOSSES (FPS)	5596.

	BOOSTERS	ORBITER
DRY WEIGHT	331368.	2008[illegible].
RESIDUAL WEIGHT	57545.	19018.
BURNOUT WEIGHT	388913.	218846.
TOTAL DRY WEIGHT	532197	
EXPENDABLES	0.	[illegible]
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		416763.
PROPELLANT WEIGHT	1779321.	914440.
WEIGHT AT LIFTOFF	2168234.	1134[illegible].
MASS FRACTION	.8206	.806[illegible]
MASS RATIO	2.87	2.78
VELOCITY THEO (FPS)	15074.	15074.
SPECIFIC IMPULSE (SEC) (VAC)	439.9	463.6
(S.L.)	392.6	375.0
(STAGE 1 AVERAGE)	443.7	
THRUST (LBF) (VAC)	4093963.	902004.
(S.L.)	3648094.	728619.
AXIAL ACCELERATION AT START	1.30	1.15
BURN TIME (SECS)	192.	256.
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	1.00	.65
NUMBER OF BOOSTERS	2.	

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	28272.	46071.
NOSE CONE	696.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	8524.	
UPPER DOME	637.	
BARREL	6168.	
LOWER DOME	1135.	
BAFFLES	584.	
INTERTANK	3470.	
SKIN	1584.	
STRINGERS	1104.	
FRAMES/BEAMS	781.	
AFT TANK	8504.	
UPPER DOME	410	
BARREL	7276.	
LOWER DOME	768.	
BAFFLES	50.	
TAIL SKIRT	7078.	
SKIN	2950.	
STRINGERS	2674.	
FRAMES	1455.	
THRUST STRUCTURE	0.	
AERO SURFACES		[illegible]
BODY		21049.
THERMAL PROTECTION SYSTEM	1417.	38565
SEPARATION	4113.	
RECOVERY	55348.	
LANDING GEAR		10041.
PROPULSION SYS	36004.	58430.
POWER SYSTEMS	4461.	
AVIONICS	2263.	
ACS WEIGHT	6105.	
ELECTRICAL	70.	5320.
I/F ATTACH	1201.	
CONTROLS		7230.
RANGE SAFETY	150.	1700.
GROWTH	27614.	33471.
INERT WEIGHT	165684.	200828.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	36004.	58430.
OXID FEED SYS	4868.	523.
FUEL FEED SYS	1595.	504.
OXID PRESS SYS	1267.	2227.
FUEL PRESS SYS	2944.	823.
PROPELLANT SYS		37837
OMS/RCS SYS		4985.
TOTAL ENGINE WEIGHT	25331.	11532.
WEIGHT OF 1 ENGINE	10415.	7032.
NUMBER OF ENGINES	2.432	1.640
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

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WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	165684.	200823
RESIDUALS	28772.	19018
GASES	8421.	6098.
LIQUIDS	2669.	2286.
ORBITER OMS PROPELLANT		10633.
FLYBACK FUEL	17015.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	889660.	914440.
OXIDIZER(FWD)	778453.	783806.
FUEL(AFT)	111208.	130634.
GROSS WEIGHT	2168234.	1134288.
RECOVERY FEATURES		
WING - TOTAL WEIGHT	36201.	
TOTAL WING AREA (SF)	3719.	
TOTAL WING SPAN (FT)	127.4	
INNER WING SPAN (FT)	51.9	
FWD ROOT CHORD (FT)	20.0	
FWD ROOT THICK (FT)	4.0	
AFT ROOT CHORD (FT)	29.2	
AFT ROOT THICK (FT)	4.4	
OUTBOARD WING SPAN (FT)	26.6	
TAIL - WEIGHT	5127.	
TAIL AREA (SF)	820.	
TAILSPAN (FT)	20.9	
TAIL CHORD (FT)	39.2	
LANDING GEAR WEIGHT	8751.	
FLYBACK ENG. WEIGHT	4419.	
THRUST (LBF)	24307.	
FLYBACK FUEL TANK WEIGHT	851.	

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 18

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PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	4409036.
THEORETICAL VELOCITY (FPS)	29819.
ACTUAL VELOCITY (FPS)	24552.
VELOCITY LOSSES (FPS)	5268.

	BOOSTERS	ORBITER
DRY WEIGHT	333165.	226718.
RESIDUAL WEIGHT	59365.	22010.
BURNOUT WEIGHT	392530.	248728
TOTAL DRY WEIGHT	559882.	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		400487.
PROPELLANT WEIGHT	2574927.	1127851.
WEIGHT AT LIFTOFF	2967457.	1376579
MASS FRACTION	.8677	.8192
MASS RATIO	3.08	3.32
VELOCITY THEO (FPS)	11928.	17892.
SPECIFIC IMPULSE (SEC) (VAC)	312.0	463.6
(S.L.)	264.5	375.0
(STAGE 1 AVERAGE)	330.0	
THRUST (LBF) (VAC)	5634161.	1180980.
(S.L.)	4776396.	955279.
AXIAL ACCELERATION AT START	1.30	1.13
BURN TIME (SEC)	152.	286.
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	.88	.65
NUMBER OF BOOSTERS	2.	

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	29902.	49280.
NOSE CONE	462.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	6261.	
UPPER DOME	467.	
BARREL	4424.	
LOWER DOME	687.	
BAFFLES	683.	
INTERTANK	4219.	
SKIN	1775.	
STRINGERS	1569.	
FRAMES/BEAMS	875.	
AFT TANK	3800.	
UPPER DOME	350.	
BARREL	2757.	
LOWER DOME	524.	
BAFFLES	169.	
TAIL SKIRT	6588.	
SKIN	2425.	
STRINGERS	2967.	
FRAMES	1196.	
THIRD TANK WEIGHT	7956.	
EXTRA INTERTANK	615.	
THRUST STRUCTURE	0.	
AERO SURFACES		20453.
BODY		22836.
THERMAL PROTECTION SYSTEM	1039.	40772.
SEPARATION	4151.	
RECOVERY	55096.	
LANDING GEAR		11336.
PROPULSION SYS	32176.	72362.
POWER SYSTEMS	5034.	
AVIONICS	2535.	
ACS WEIGHT	8867.	
ELECTRICAL	61.	5320.
I/F ATTACH	1738.	
CONTROLS		8162.
RANGE SAFETY	150.	1700.
GROWTH	27764.	37786.
INERT WEIGHT	166582.	226718.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	32176.	72362.
OXID FEED SYS	4125.	1078.
FUEL FEED SYS	511.	268.
OXID PRESS SYS	1632.	2746.
FUEL PRESS SYS	2208.	1015.
PROPELLANT SYS		46667.
OMS/RCS SYS		5490.
TOTAL ENGINE WEIGHT	23700.	15098.
WEIGHT OF 1 ENGINE	7443.	7032.
NUMBER OF ENGINES	3.184	2.147
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

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WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	166582.	226718.
RESIDUALS	29683.	22010
GASES	7681.	75[illegible]
LIQUIDS	3862.	3850.
ORBITER OMS PROPELLANT		11669.
FLYBACK FUEL	17173.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1287464.	11[illegible]1.
OXIDIZER(FWD)	911013.	9[illegible]28.
FUEL(AFT)	376451.	1611[illegible].
GROSS WEIGHT	2967457.	1376579.
RECOVERY FEATURES		
WING - TOTAL WEIGHT	35770.	
TOTAL WING AREA (SF)	3754.	
TOTAL WING SPAN (FT)	122.9	
INNER WING SPAN (FT)	48.8	
FWD ROOT CHORD (FT)	30.0	
FWD ROOT THICK (FT)	4.0	
AFT ROOT CHORD (FT)	30.6	
AFT ROOT THICK (FT)	4.6	
OUTBOARD WING SPAN (FT)	27.9	
TAIL - WEIGHT	5175.	
TAIL AREA (SF)	828.	
TAILSPAN (FT)	21.0	
TAIL CHORD (FT)	39.4	
LANDING GEAR WEIGHT	8832.	
FLYBACK ENG. WEIGHT	4461.	
THRUST (LBF)	24533.	
FLYBACK FUEL TANK WEIGHT	859.	

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 25

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PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3880068.
THEORETICAL VELOCITY (FPS)	30291.
ACTUAL VELOCITY (FPS)	24552.
VELOCITY LOSSES (FPS)	5739.

	BOOSTERS		ORBITER
DRY WEIGHT	307699		167850
RESIDUAL WEIGHT	49625.		15072
BURNOUT WEIGHT	357324.		182922.
TOTAL DRY WEIGHT		475549.	
EXPENDABLES	0.		0.
SHROUD WEIGHT	0.		
PARALLEL BURNED PROP			214286.
PROPELLANT WEIGHT	2624081.		650739.
WEIGHT AT LIFTOFF	2981405.		833661.
MASS FRACTION	.8801		.7806
MASS RATIO	3.72		[illegible]
VELOCITY THEO (FPS)	15145.		15145.
SPECIFIC IMPULSE (SEC) (VAC)	350.0		463.6
(S.L.)	306.3		375.0
(STAGE 1 AVERAGE)		358.0	
THRUST (LBF) (VAC)	5239699.		566888.
(S.L.)	4585485.		458549.
AXIAL ACCELERATION AT START	1.30		.83
BURN TIME (SECS)	205.		257.
THROTTLE RATIO AT MAX Q	1.00		1.00
AT BOOSTER B.O.	.72		.65
NUMBER OF BOOSTERS	2.		

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	30150.	41513.
NOSE CONE	502.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	6245.	
UPPER DOME	428.	
BARREL	4386.	
LOWER DOME	689.	
BAFFLES	741.	
INTERTANK	3185.	
SKIN	1362.	
STRINGERS	1153.	
FRAMES/BEAMS	671.	
AFT TANK	5405.	
UPPER DOME.	320.	
BARREL	4364.	
LOWER DOME	575.	
BAFFLES	146.	
TAIL SKIRT	6826.	
SKIN	2582.	
STRINGERS	2970.	
FRAMES	1273.	
THIRD TANK WEIGHT	7330.	
EXTRA INTERTANK	657.	
THRUST STRUCTURE	0.	
AERO SURFACES		22947
BODY		18566.
THERMAL PROTECTION SYSTEM	1141.	35430
SEPARATION	3779.	
RECOVERY	50558.	
LANDING GEAR		8383.
PROPULSION SYS	27538.	41327.
POWER SYSTEMS	5111.	
AVIONICS	2539.	
ACS WEIGHT	7375.	
ELECTRICAL	64.	5320.
I/F ATTACH	1771.	
CONTROLS		6055.
RANGE SAFETY	150.	1700.
GROWTH	25642.	27942.
INERT WEIGHT	153849.	167650.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	27538.	41327.
OXID FEED SYS	3734.	517.
FUEL FEED SYS	[illegible]	[illegible]
OXID PRESS SYS	740.	[illegible]
FUEL PRESS SYS	1390.	[illegible]
PROPELLANT SYS		26925.
OMS/RCS SYS		[illegible]
TOTAL ENGINE WEIGHT	21054.	[illegible]
WEIGHT OF 1 ENGINE	6887.	[illegible]
NUMBER OF ENGINES	3.057	[illegible]
OPERATING THRUST (LBF)	750000.	[illegible]
THRUST LEVEL	1.000	[illegible]

WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	153849.	167650.
RESIDUALS	24813.	15273.
GASES	4259.	4340.
LIQUIDS	3936.	1627
ORBITER OMS PROPELLANT		9306.
FLYBACK FUEL	15633.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1312041.	650740.
OXIDIZER(FWD)	988080.	557777.
FUEL(AFT)	323961.	92963.
GROSS WEIGHT	2981405.	833662.
RECOVERY FEATURES		
WING - TOTAL WEIGHT	32968.	
TOTAL WING AREA (SF)	3417.	
TOTAL WING SPAN (FT)	119.6	
INNER WING SPAN (FT)	49.5	
FWD ROOT CHORD (FT)	20.0	
FWD ROOT THICK (FT)	4.0	
AFT ROOT CHORD (FT)	27.7	
AFT ROOT THICK (FT)	4.2	
OUTBOARD WING SPAN (FT)	25.3	
TAIL - WEIGHT	4710.	
TAIL AREA (SF)	754.	
TAILSPAN (FT)	20.0	
TAIL CHORD (FT)	37.6	
LANDING GEAR WEIGHT	8040.	
FLYBACK ENG. WEIGHT	4061.	
THRUST (LBF)	22333.	
FLYBACK FUEL TANK WEIGHT	782.	

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 29

PAYLOAD WEIGHT	65000.	
GROSS LIFT-OFF WEIGHT	3948339.	
THEORETICAL VELOCITY (FPS)	29974.	
ACTUAL VELOCITY (FPS)	2455[illegible].	
VELOCITY LOSSES (FPS)	542[illegible].	

	BOOSTERS	ORBITER
DRY WEIGHT	280205.	220640.
RESIDUAL WEIGHT	42844.	21407
BURNOUT WEIGHT	323049.	242047.
TOTAL DRY WEIGHT	500845.	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		370372.
PROPELLANT WEIGHT	[illegible]9636.	1083608
WEIGHT AT LIFTOFF	2552685.	1330654.
MASS FRACTION	.8734	.8181
MASS RATIO	2.93	3.34
VELOCITY THEO (FPS)	11990.	17984.
SPECIFIC IMPULSE (SECS) (VAC)	330.2	463.6
(S.L.)	287.0	375.0
(STAGE 1 AVERAGE)	346.8	
THRUST (LBF) (VAC)	4921161.	1057583.
(S.L.)	4277326.	855465.
AXIAL ACCELERATION AT START	1.30	1.03
BURN TIME (SECS)	153.	315.
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	.96	.65
NUMBER OF BOOSTERS	2.	

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	25703.	48716.
NOSE CONE	417.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	5157.	
UPPER DOME	365.	
BARREL	3642.	
LOWER DOME	544.	
BAFFLES	605.	
INTERTANK	2769.	
SKIN	1158.	
STRINGERS	1039.	
FRAMES/BEAMS	571.	
AFT TANK	4249.	
UPPER DOME.	265.	
BARREL	2404.	
LOWER DOME	441.	
BAFFLES	139.	
TAIL SKIRT	5611.	
SKIN	2068.	
STRINGERS	2523.	
FRAMES	1020.	
THIRD TANK WEIGHT	6964.	
EXTRA INTERTANK	538.	
THRUST STRUCTURE	0.	
AERO SURFACES		26202.
BODY		22514
THERMAL PROTECTION SYSTEM	963.	40284.
SEPARATION	3416	
RECOVERY	45648.	
LANDING GEAR		11032.
PROPULSION SYS	25806.	68772.
POWER SYSTEMS	4641.	
AVIONICS	2401.	
ACS WEIGHT	8132.	
ELECTRICAL	59.	5320.
I/F ATTACH	1505.	
CONTROLS		7945.
RANGE SAFETY	150.	1700.
GROWTH	23350.	36773.
INERT WEIGHT	140103.	220640.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	25806.	63772.
OXID FEED SYS	3653.	965.
FUEL FEED SYS	543.	340.
OXID PRESS SYS	559.	1661.
FUEL PRESS SYS	995.	378.
PROPELLANT SYS		45043.
OMS/RCS SYS		5373.
TOTAL ENGINE WEIGHT	20056.	13521.
WEIGHT OF 1 ENGINE	7033.	7032.
NUMBER OF ENGINES	2.852	1.923
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	140103.	220640.
RESIDUALS	11423.	[illegible]
GASES	3108.	[illegible]
LIQUIDS	3344.	[illegible]
ORBITER OMS PROPELLANT		11428.
FLYBACK FUEL	14133.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1114818.	[illegible]
OXIDIZER(FWD)	806857.	933091.
FUEL(AFT)	307961.	155515.
GROSS WEIGHT	2552685.	1330654.
RECOVERY FEATURES		
WING - TOTAL WEIGHT	29743.	
TOTAL WING AREA (SF)	3089.	
TOTAL WING SPAN (FT)	112.4	
INNER WING SPAN (FT)	47.9	
FWD ROOT CHORD (FT)	20.0	
FWD ROOT THICK (FT)	4.0	
AFT ROOT CHORD (FT)	25.7	
AFT ROOT THICK (FT)	3.8	
OUTBOARD WING SPAN (FT)	23.4	
TAIL - WEIGHT	4259.	
TAIL AREA (SF)	681.	
TAILSPAN (FT)	19.1	
TAIL CHORD (FT)	35.7	
LANDING GEAR WEIGHT	7269.	
FLYBACK ENG. WEIGHT	3671.	
THRUST (LBF)	20191.	
FLYBACK FUEL TANK WEIGHT	707.	

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PERFORMANCE PARAMETERS

NUMBER OF ITERATIONS 24

PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3904012.
THEORETICAL VELOCITY (FPS)	30011.
ACTUAL VELOCITY (FPS)	24552.
VELOCITY LOSSES (FPS)	5460.

	BOOSTERS		ORBITER
DRY WEIGHT	279127.		193390.
RESIDUAL WEIGHT	43019.		18179.
BURNOUT WEIGHT	322146.		211569.
TOTAL DRY WEIGHT		472517.	
EXPENDABLES	0.		0.
SHROUD WEIGHT	0.		
PARALLEL BURNED PROP			295631.
PROPELLANT WEIGHT	2449852.		855444
WEIGHT AT LIFTOFF	2771998.		1067014
MASS FRACTION	.8838		.8017
MASS RATIO	3.37		3.02
VELOCITY THEO (FPS)	13505.		16506.
SPECIFIC IMPULSE (SECS) (VAC)	332.8		463.6
(S.L.)	290.7		375.0
(STAGE 1 AVERAGE)		345.5	
THRUST (LBF) (VAC)	5052356.		818388.
(S.L.)	4413221.		661983.
AXIAL ACCELERATION AT START	1.30		.98
BURN TIME (SECS)	179.		317.
THROTTLE RATIO AT MAX Q	1.00		1.00
AT BOOSTER B.O.	.81		.65
NUMBER OF BOOSTERS	2.		

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SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	27129.	45116.
NOSE CONE	417.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	5323.	
UPPER DOME	367.	
BARREL	3739	
LOWER DOME	547.	
BAFFLES	670.	
INTERTANK	2821.	
SKIN	1174.	
STRINGERS	1067.	
FRAMES/BEAMS	579.	
AFT TANK	3763.	
UPPER DOME.	290.	
BARREL	2877.	
LOWER DOME	447.	
BAFFLES	149.	
TAIL SKIRT	5746.	
SKIN	2104.	
STRINGERS	2605.	
FRAMES	1037.	
THIRD TANK WEIGHT	8511.	
EXTRA INTERTANK	548.	
THRUST STRUCTURE	0.	
AERO SURFACES		24596
BODY		20520.
THERMAL PROTECTION SYSTEM	962.	37806.
SEPARATION	3407.	
RECOVERY	45533.	
LANDING GEAR		9670.
PROPULSION SYS	23459.	54483.
POWER SYSTEMS	4835.	
AVIONICS	2473.	
ACS WEIGHT	8480.	
ELECTRICAL	59.	5520.
I/F ATTACH	1654.	
CONTROLS		6962.
RANGE SAFETY	150.	1700.
GROWTH	23261.	32133.
INERT WEIGHT	139564.	193390.

PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	23459.	54483.
OXID FEED SYS	3061.	747.
FUEL FEED SYS	397.	186.
OXID PRESS SYS	560.	2083.
FUEL PRESS SYS	851.	770.
PROPELLANT SYS		35395.
OMS/RCS SYS		4340.
TOTAL ENGINE WEIGHT	18590.	10463.
WEIGHT OF 1 ENGINE	6318.	7032.
NUMBER OF ENGINES	2.942	1.488
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

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WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	139564.	193380.
RESIDUALS	21509.	[illegible]
GASES	2822.	[illegible]
LIQUIDS	3675.	[illegible]
ORBITER OMS PROPELLANT		[illegible]
FLYBACK FUEL	14094.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1224926.	855444.
OXIDIZER(FWD)	893865.	733238.
FUEL(AFT)	331061.	122206
GROSS WEIGHT	2771998.	1067014.

RECOVERY FEATURES

WING - TOTAL WEIGHT	39672.
TOTAL WING AREA (SF)	3081.
TOTAL WING SPAN (FT)	112.5
INNER WING SPAN (FT)	47.9
FWD ROOT CHORD (FT)	20.0
FWD ROOT THICK (FT)	4.0
AFT ROOT CHORD (FT)	25.6
AFT ROOT THICK (FT)	3.8
OUTBOARD WING SPAN (FT)	23.3
TAIL - WEIGHT	4247.
TAIL AREA (SF)	679.
TAILSPAN (FT)	19.0
TAIL CHORD (FT)	35.7
LANDING GEAR WEIGHT	7248.
FLYBACK ENG. WEIGHT	3661.
THRUST (LBF)	20134.
FLYBACK FUEL TANK WEIGHT	705.

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 28

PAYLOAD WEIGHT	65000.	
GROSS LIFT-OFF WEIGHT	3959405.	
THEORETICAL VELOCITY (FPS)	30036.	
ACTUAL VELOCITY (FPS)	24552.	
VELOCITY LOSSES (FPS)	5484.	

	BOOSTERS	ORBITER
DRY WEIGHT	290480.	221601.
RESIDUAL WEIGHT	47080.	21533.
BURNOUT WEIGHT	337560.	243134.
TOTAL DRY WEIGHT	512081.	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		374815.
PROPELLANT WEIGHT	2215586.	1098125.
WEIGHT AT LIFTOFF	2553146.	1341259.
MASS FRACTION	.8678	.8187
MASS RATIO	2.89	3.35
VELOCITY THEO (FPS)	12014.	18021.
SPECIFIC IMPULSE (SECS) (VAC)	335.4	463.6
(S.L.)	294.6	375.0
(STAGE 1 AVERAGE)	351.6	
THRUST (LBF) (VAC)	4883339.	1060544.
(S.L.)	4289301.	857360.
AXIAL ACCELERATION AT START	1.30	1.03
BURN TIME (SECS)	154.	316.
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	.98	.65
NUMBER OF BOOSTERS	2.	

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SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	27980.	48853.
NOSE CONE	462.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	4997.	
UPPER DOME	282.	
BARREL	4218.	
LOWER DOME	282.	
BAFFLES	215.	
INTERTANK	3900.	
SKIN	1650.	
STRINGERS	1437.	
FRAMES/BEAMS	814.	
AFT TANK	1010.	
UPPER DOME	334.	
BARREL	261.	
LOWER DOME	409.	
BAFFLES	6.	
INTERTANK	2497.	
SKIN	1064.	
STRINGERS	908.	
FRAMES/BEAMS	525.	
THIRD TANK	9213.	
UPPER DOME	880.	
BARREL	6848.	
LOWER DOME	1121.	
BAFFLES	364.	
TAIL SKIRT	5901.	
SKIN	2204.	
STRINGERS	2610.	
FRAMES	1087.	
THRUST STRUCTURE	0.	
AERO SURFACES		26263.
BODY		22590.
THERMAL PROTECTION SYSTEM	1038.	40479.
SEPARATION	3570.	
RECOVERY	47681.	
LANDING GEAR		11080.
PROPULSION SYS	25475.	69258.
POWER SYSTEMS	4692.	
AVIONICS	2401.	
ACS WEIGHT	8151.	
ELECTRICAL	62.	5320.
I/F ATTACH	1496.	
CONTROLS		7978.
RANGE SAFETY	150.	1700.
GROWTH	24207.	36934.
INERT WEIGHT	145240.	221601.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	25475.	69258.
OXID FEED SYS	3260.	963.
FUEL FEED SYS	1483.	240.
OXID PRESS SYS	673.	2674.
FUEL PRESS SYS	319.	[illegible]
SECOND FUEL PRES SYS	1316.	
SECOND FUEL FEED SYS	521.	
PROPELLANT SYS		48437.
OMS/RCS SYS		5392.
TOTAL ENGINE WEIGHT	17902.	13559.
WEIGHT OF 1 ENGINE	6260.	7032.
NUMBER OF ENGINES	2.860	1.928
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	145240.	221801.
RESIDUALS	13540.	11583.
GASES	4617.	[illegible]
LIQUIDS	3323.	[illegible]745.
ORBITER OMS PROPELLANT		11464.
FLYBACK FUEL	14768.	
EXPENDABLES	0.	0
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1107793	[illegible]
OXIDIZER(AFT)	808799.	941050
FUEL(FWD)	286808.	156675
SECOND FUEL(MID)	12186.	
GROSS WEIGHT	2553146.	1341259.
RECOVERY FEATURES		
WING - TOTAL WEIGHT	31062.	
TOTAL WING AREA (SF)	3228.	
TOTAL WING SPAN (FT)	115.2	
INNER WING SPAN (FT)	48.5	
FWD ROOT CHORD (FT)	20.0	
FWD ROOT THICK (FT)	4.0	
AFT ROOT CHORD (FT)	26.6	
AFT ROOT THICK (FT)	4.0	
OUTBOARD WING SPAN (FT)	24.3	
TAIL - WEIGHT	4450.	
TAIL AREA (SF)	712.	
TAILSPAN (FT)	19.5	
TAIL CHORD (FT)	36.5	
LANDING GEAR WEIGHT	7595.	
FLYBACK ENG. WEIGHT	3836.	
THRUST (LBF)	21097.	
FLYBACK FUEL TANK WEIGHT	738.	

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 24

PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3805624.
THEORETICAL VELOCITY (FPS)	30274.
ACTUAL VELOCITY (FPS)	24552.
VELOCITY LOSSES (FPS)	5723.

	BOOSTERS		ORBITER
DRY WEIGHT	299395.		221853.
RESIDUAL WEIGHT	48162.		21643.
BURNOUT WEIGHT	347557.		243496
TOTAL DRY WEIGHT		521248.	
EXPENDABLES	0.		0.
SHROUD WEIGHT	0.		
PARALLEL BURNED PROP			374923.
PROPELLANT WEIGHT	2040508.		1109084
WEIGHT AT LIFTOFF	2388065.		1552560
MASS FRACTION	.8545		.8[illegible]
MASS RATIO	2.74		[illegible]
VELOCITY THEO (FPS)	12110		18165
SPECIFIC IMPULSE (SECS) (VAC)	359.8		463.6
(S.L.)	316.2		375.0
(STAGE 1 AVERAGE)		373.7	
THRUST (LBF) (VAC)	4729334.		1019352.
(S.L.)	4122703.		824541.
AXIAL ACCELERATION AT START	1.30		.88
BURN TIME (SECS)	156.		334.
THROTTLE RATIO AT MAX Q	1.00		1.00
AT BOOSTER B.O.	1.00		.65
NUMBER OF BOOSTERS	2.		

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	30063.	49011.
NOSE CONE	573.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	5905.	
UPPER DOME	317.	
BARREL	5025.	
LOWER DOME	396.	
BAFFLES	167.	
INTERTANK	5173.	
SKIN	2251.	
STRINGERS	1812.	
FRAMES/BEAMS	1110.	
AFT TANK	907.	
UPPER DOME.	412.	
BARREL	0.	
LOWER DOME	490.	
BAFFLES	5.	
INTERTANK	2689.	
SKIN	1177.	
STRINGERS	931.	
FRAMES/BEAMS	581.	
THIRD TANK	8525.	
UPPER DOME.	1064.	
BARREL	5789.	
LOWER DOME	1318.	
BAFFLES	254.	
TAIL SKIRT	6291.	
SKIN	2431.	
STRINGERS	2661.	
FRAMES	1199.	
THRUST STRUCTURE	0.	
AERO SURFACES		26333.
BODY		22678.
THERMAL PROTECTION SYSTEM	1217.	40587.
SEPARATION	3675.	
RECOVERY	49251.	
LANDING GEAR		11093.
PROPULSION SYS	27058.	69180.
POWER SYSTEMS	4654.	
AVIONICS	2344.	
ACS WEIGHT	6420.	
ELECTRICAL	68.	5320.
I/F ATTACH	1377.	
CONTROLS		7987.
RANGE SAFETY	150.	1700.
GROWTH	24950.	36975.
INERT WEIGHT	149698.	221853.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	27058.	69180
OXID FEED SYS	4511.	430.
FUEL FEED SYS	1166.	[illegible]1.
OXID PRESS SYS	956.	[illegible]
FUEL PRESS SYS	289.	[illegible]
SECOND FUEL PRES SYS	1280.	
SECOND FUEL FEED SYS	523.	
PROPELLANT SYS		[illegible]
OMS/RCS SYS		[illegible]
TOTAL ENGINE WEIGHT	18334.	[illegible]
WEIGHT OF 1 ENGINE	6671.	703[illegible].
NUMBER OF ENGINES	2.748	1.853
OPERATING THRUST (LBF)	750000.	650000.
THRUST LEVEL	1.000	1.000

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WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	149698.	221853.
RESIDUALS	24081.	[illegible]
GASES	5049.	[illegible]
LIQUIDS	3061.	[illegible]
ORBITER OMS PROPELLANT		11474
FLYBACK FUEL	15206.	
EXPENDABLES	0.	0
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1020254.	1109064.
OXIDIZER(AFT)	786446.	950628.
FUEL(FWD)	222789.	158436.
SECOND FUEL(MID)	11019.	
GROSS WEIGHT	[illegible]	[illegible]

RECOVERY FEATURES

WING - TOTAL WEIGHT	32139.
TOTAL WING AREA (SF)	3024.
TOTAL WING SPAN (FT)	118.0
INNER WING SPAN (FT)	49.5
FWD ROOT CHORD (FT)	20.0
FWD ROOT THICK (FT)	4.0
AFT ROOT CHORD (FT)	27.0
AFT ROOT THICK (FT)	4.1
OUTBOARD WING SPAN (FT)	24.6
TAIL - WEIGHT	4583.
TAIL AREA (SF)	733.
TAILSPAN (FT)	19.8
TAIL CHORD (FT)	37.1
LANDING GEAR WEIGHT	7820.
FLYBACK ENG. WEIGHT	3950.
THRUST (LBF)	21722.
FLYBACK FUEL TANK WEIGHT	760.

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 25

PAYLOAD WEIGHT	65000.
GROSS LIFT-OFF WEIGHT	3914412.
THEORETICAL VELOCITY (FPS)	30108.
ACTUAL VELOCITY (FPS)	24552.
VELOCITY LOSSES (FPS)	5556

	BOOSTERS	ORBITER
DRY WEIGHT	300983.	221882.
RESIDUAL WEIGHT	48399.	21593
BURNOUT WEIGHT	349383.	243475.
TOTAL DRY WEIGHT	522865.	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		376357.
PROPELLANT WEIGHT	2153058.	[illegible]
WEIGHT AT LIFTOFF	2502441.	1346871.
MASS FRACTION	.8604	.819[illegible]
MASS RATIO	2.83	3.[illegible]
VELOCITY THEO (FPS)	12043.	18065.
SPECIFIC IMPULSE (SEC) (VAC)	344.9	463.6
(S.L.)	302.3	375.0
(STAGE 1 AVERAGE)	360.3	
THRUST (LBF) (VAC)	4833891.	1048506.
(S.L.)	4240616.	848123.
AXIAL ACCELERATION AT START	1.30	1.01
BURN TIME (SECS)	154.	322
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	.99	.65
NUMBER OF BOOSTERS	2.	

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	29329.	48931.
NOSE CONE	525.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	5547.	
UPPER DOME	304.	
BARREL	4732.	
LOWER DOME	318.	
BAFFLES	193.	
INTERTANK	4735.	
SKIN	2034.	
STRINGERS	1698.	
FRAMES/BEAMS	1003.	
AFT TANK	667.	
UPPER DOME.	383.	
BARREL	20	
LOWER DOME	45[illegible].	
BAFFLES	5.	
INTERTANK	26[illegible].	
SKIN	1139.	
STRINGERS	930.	
FRAMES/BEAMS	562.	
THIRD TANK	8818.	
UPPER DOME.	991.	
BARREL	6219.	
LOWER DOME	1244.	
BAFFLES	363.	
TAIL SKIRT	6205.	
SKIN	2362.	
STRINGERS	2679.	
FRAMES	1165.	
THRUST STRUCTURE	0.	
AERO SURFACES		26298.
BODY		22633
THERMAL PROTECTION SYSTEM	1143.	40532.
SEPARATION	3695.	
RECOVERY	49298.	
LANDING GEAR		11094.
PROPULSION SYS	26609.	69336.
POWER SYSTEMS	4715.	
AVIONICS	2384.	
ACS WEIGHT	8083.	
ELECTRICAL	66.	5320.
I/F ATTACH	1453.	
CONTROLS		7988.
RANGE SAFETY	150.	1700.
GROWTH	25082.	36980.
INERT WEIGHT	150492.	221882.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	26609.	69336.
OXID FEED SYS	3999.	367.
FUEL FEED SYS	1262.	336.
OXID PRESS SYS	834.	2687
FUEL PRESS SYS	391.	993.
SECOND FUEL PRES SYS	1314.	
SECOND FUEL FEED SYS	536.	
PROPELLANT SYS		45659.
OMS/RCS SYS		5398.
TOTAL ENGINE WEIGHT	18374.	13405.
WEIGHT OF 1 ENGINE	6499.	7032.
NUMBER OF ENGINES	2.827	1.906
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	1.000

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WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	150492.	221882.
RESIDUALS	24200.	[illegible]
GASES	4877.	7050.
LIQUIDS	3230.	2758.
ORBITER OMS PROPELLANT		11475.
FLYBACK FUEL	15285.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1076529.	[illegible]
OXIDIZER(AFT)	807465.	945854.
FUEL(FWD)	257976.	15764[illegible].
SECOND FUEL(MID)	11088.	
GROSS WEIGHT	2502441.	1348971.

RECOVERY FEATURES

WING - TOTAL WEIGHT	32197.
TOTAL WING AREA (SF)	3341.
TOTAL WING SPAN (FT)	117.8
INNER WING SPAN (FT)	49.1
FWD ROOT CHORD (FT)	20.0
FWD ROOT THICK (FT)	4.0
AFT ROOT CHORD (FT)	27.3
AFT ROOT THICK (FT)	4.1
OUTBOARD WING SPAN (FT)	34.9
TAIL - WEIGHT	4606.
TAIL AREA (SF)	737.
TAILSPAN (FT)	19.8
TAIL CHORD (FT)	37.2
LANDING GEAR WEIGHT	7861.
FLYBACK ENG. WEIGHT	3970.
THRUST (LBF)	21836.
FLYBACK FUEL TANK WEIGHT	764.

SP/H UFRCV

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PERFORMANCE PARAMETERS
NUMBER OF ITERATIONS 27

PAYLOAD WEIGHT	65000.	
GROSS LIFT-OFF WEIGHT	4097657.	
THEORETICAL VELOCITY (FPS)	30187.	
ACTUAL VELOCITY (FPS)	24551.	
VELOCITY LOSSES (FPS)	564[illegible].	

	BOOSTERS	[illegible]
DRY WEIGHT	336856.	169379.
RESIDUAL WEIGHT	56676.	15440.
BURNOUT WEIGHT	393532.	184[illegible].
TOTAL DRY WEIGHT	[illegible].	
EXPENDABLES	0.	0.
SHROUD WEIGHT	0.	
PARALLEL BURNED PROP		[illegible]
PROPELLANT WEIGHT	2792900.	[illegible]
WEIGHT AT LIFTOFF	3186432.	846[illegible]4.
MASS FRACTION	.8765	.7816
MASS RATIO	3.79	[illegible]
VELOCITY THEO (FPS)	15099.	[illegible]
SPECIFIC IMPULSE (SEC)(VAC)	343.9	463.6
(S.L.)	301.6	375.0
(STAGE 1 AVERAGE)	352.2	
THRUST (LBF) (VAC)	5522047.	5987[illegible].
(S.L.)	4842830.	484[illegible].
AXIAL ACCELERATION AT START	1.30	.87
BURN TIME (SECS)	205.	[illegible].
THROTTLE RATIO AT MAX Q	1.00	1.00
AT BOOSTER B.O.	.71	.65
NUMBER OF BOOSTERS	2.	

SUBSYSTEMS WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
STRUCTURE	33478.	41713.
NOSE CONE	540.	
FORWARD NONTANK	0.	
SKIN	0.	
STRINGERS	0.	
FRAMES	0.	
FORWARD TANK	6215.	
UPPER DOME	323.	
BARREL	5315.	
LOWER DOME	326.	
BAFFLES	251.	
INTERTANK	3395.	
SKIN	1439.	
STRINGERS	1246.	
FRAMES/BEAMS	710.	
AFT TANK	1167.	
UPPER DOME.	402.	
BARREL	261.	
LOWER DOME	497.	
BAFFLES	6.	
INTERTANK	3021.	
SKIN	1291.	
STRINGERS	1092.	
FRAMES/BEAMS	637.	
THIRD TANK	11989.	
UPPER DOME.	1060.	
BARREL	9065.	
LOWER DOME	1393.	
BAFFLES	471.	
TAIL SKIRT	7152.	
SKIN	2679.	
STRINGERS	3152.	
FRAMES	1321.	
THRUST STRUCTURE	0.	
AERO SURFACES		23077.
BODY		18637.
THERMAL PROTECTION SYSTEM	1171.	35563.
SEPARATION	4162.	
RECOVERY	55411.	
LANDING GEAR		8469.
PROPULSION SYS	29010.	42281.
POWER SYSTEMS	5313.	
AVIONICS	2601.	
ACS WEIGHT	9206.	
ELECTRICAL	65.	5320.
I/F ATTACH	1885.	
CONTROLS		6098.
RANGE SAFETY	150.	1700.
GROWTH	28071.	28230.
INERT WEIGHT	168428.	169379.

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PROPULSION SYSTEM WEIGHT SUMMARY

	SINGLE BOOSTER	ORBITER
PROPULSION SYS WT	29010.	[illegible]
OXID FEED SYS	3661.	[illegible]
FUEL FEED SYS	1719.	136.
OXID PRESS SYS	861.	[illegible]
FUEL PRESS SYS	377.	596.
SECOND FUEL PRES SYS	1705.	
SECOND FUEL FEED SYS	652.	
PROPELLANT SYS		[illegible]
OMS/RCS SYS		[illegible]
TOTAL ENGINE WEIGHT	20056.	7654
WEIGHT OF 1 ENGINE	6212.	[illegible]
NUMBER OF ENGINES	3.229	[illegible]
OPERATING THRUST (LBF)	750000.	550000.
THRUST LEVEL	1.000	[illegible]

WEIGHT SUMMARY -CONT.

	BOOSTER	STAGE 2
DRY WEIGHT	168428.	16[illegible].
RESIDUALS	28338.	15440.
GASES	5884.	4411.
LIQUIDS	4189.	1654.
ORBITER OMS PROPELLANT		9375.
FLYBACK FUEL	17217.	
EXPENDABLES	0.	0.
THRUST VECTOR CONTROL	0.	
USABLE PROPELLANT	1396450.	661406.
OXIDIZER(AFT)	1047426.	566320.
FUEL(FWD)	334641.	94487.
SECOND FUEL(MID)	14383.	
GROSS WEIGHT	3186432.	846[illegible].

RECOVERY FEATURES

WING - TOTAL WEIGHT	36036.
TOTAL WING AREA (SF)	3763.
TOTAL WING SPAN (FT)	124.3
INNER WING SPAN (FT)	49.6
FWD ROOT CHORD (FT)	20.0
FWD ROOT THICK (FT)	4.0
AFT ROOT CHORD (FT)	30.4
AFT ROOT THICK (FT)	4.6
OUTBOARD WING SPAN (FT)	37.7
TAIL - WEIGHT	5188.
TAIL AREA (SF)	830.
TAILSPAN (FT)	21.0
TAIL CHORD (FT)	39.5
LANDING GEAR WEIGHT	8854.
FLYBACK ENG. WEIGHT	4472.
THRUST (LBF)	24596.
FLYBACK FUEL TANK WEIGHT	861.

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Glossary

3-DOpt	Three-dimensional Optimal Flight Trajectory Program
AFB	Air Force Base
AFRPL	Air Force Rocket Propulsion Laboratory
ALS	Advanced Launch Systems
APS	Auxiliary Propulsion System
APU	Auxiliary Power Unit
ASTM	American Society for Testing Materials
Bfrac	Variable for UFRCV representing fraction of booster ideal velocity supplied when either a VMRE is operating in mode one, or a translating nozzle booster engine is operating at a low expansion ratio
BTNE	Booster Translating Nozzle Engine
C3H8	Chemical formula for propane
CH4	Methane
cm	Centimeter
EFF	Pump Efficiency factor
ELES	Expendable Liquid Engine Simulation
ETR	Eastern Test Range
F/L	Fine, or slow, Fill
FLYIT	Name for trajectory analysis program
g	acceleration in multiples of acceleration due to gravity
Gg	Giga-grams, (billion)
GH2	Gaseous Hydrogen
GHE or GHe	Gaseous Helium
GLOM	Gross Liftoff Mass
GN2	Gaseous Nitrogen
GOX	Gaseous Oxygen
GSS	Ground Support System
H/H	Engine with hydrogen as fuel and coolant
H2	Hydrogen
HC	Hydrocarbon
HMRE	High Mixture Ratio Engine
Isp	Specific Impulse

J/Min	Joules per minute
J/Mol	Joules per mole
Kg	Kilogram
Km	Kilometer
KN	KiloNewton
KPa	KiloPascal
KSC	Kennedy Space Center
KW	Kilowatts
L	Liter
L/D	Length to Diameter Ratio for pipe
LH2	Liquid Hydrogen
LN2	Liquid Nitrogen
LO2	Liquid Oxygen
LOX	Liquid Oxygen
LPM	Liters per minute
m	Meter
M/H	Engine with methane as fuel, hydrogen as coolant
M/M	Engine with methane as fuel and coolant
Mg	Million grams
MIPROPS	Interactive FORTRAN Programs for Micro Computers to Calculate the Thermophysical Properties of Twelve Fluids
MN	Million Newtons
MPa	Million Pascals
mps	meters per second
NASA	National Aeronautics and Space Administration
NBP	Normal Boiling Point
NP/H	Engine with NBP propane as fuel, hydrogen as coolant
NP/NP	Engine with NBP propane as fuel and coolant
NPSH	Net Pressure Suction Head
NSTL	National Space Technology Laboratory
O2	Oxygen
OMS	Orbital Maneuvering System
PB1	Variable representing the fraction of boost phase of an SSTO during which the VMRE is operating in mode one (high mixture ratio)

PB1Frac	Variable representing the fraction of boost phase of an SSTO during which a translating nozzle is operating at a low expansion ratio
PC	Personal Computer
POST	Program to Optimize Simulated Trajectories
R/H	Engine with RP-1 as fuel, and hydrogen as coolant
R/L	Rapid, or fast, fill
R/R	Engine with RP-1 as fuel and coolant
RCS	Reaction Control System
ROM	Rough Order of Magnitude
RP-1	Designation for a hydrocarbon rocket fuel
Sc	Subcooled
SDV	Shuttle Derived Vehicle
Sp. Gr.	Specific Gravity
SP/H	Engine with subcooled propane as fuel, hydrogen as coolant
SP/SP	Engine with subcooled propane as fuel and coolant
SSME	Space Shuttle Main Engine
SSTO	Single-Stage-to-Orbit
STAS	Space Transportation Architectural Study
STBE	Space Transportation Booster Engine
STME	Space Transportation Main Engine
STS	Space Transportation System
Tfrac	Fraction of thrust provided by boost engines in an SSTO
UFRCV	Unmanned Fully Reusable Cargo Vehicle
VAFB	Vandenberg Air Force Base
VMRE	Variable Mixture Ratio Engine
WASP	Weight and Sizing Program
WTNOZ	extendable nozzle weight
XFR	Transfer

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16. Abstract
THIS DOCUMENT IS THE FINAL REPORT FOR A STUDY EXAMINING THE IMPACT ON LAUNCH VEHICLES FOR VARIOUS BOOST PROPULSION DESIGN OPTIONS. THESE OPTIONS INCLUDED: DIFFERENT BOOST PHASE ENGINES USING DIFFERENT COMBINATIONS OF FUELS AND COOLANTS TO INCLUDE RP-1, METHANE, PROPANE (SUBCOOLED AND NORMAL BOILING POINT), AND HYDROGEN; VARIABLE AND HIGH MIXTURE RATIO HYDROGEN ENGINES; TRANSLATING NOZZLES ON BOOST PHASE ENGINES; AND CROSS FEEDING PROPELLANTS FROM THE BOOSTER TO SECOND STAGE. VEHICLES EXAMINED INCLUDED A FULLY REUSABLE TWO STAGE CARGO VEHICLE AND A SINGLE STAGE TO ORBIT VEHICLE. THE USE OF SUBCOOLED PROPANE AS A FUEL GENERATED VEHICLES WITH THE LOWEST TOTAL VEHICLE DRY MASS. ENGINES WITH HYDROGEN COOLING GENERATED ONLY SLIGHT MASS REDUCTIONS FROM THE REFERENCE, ALL HYDROGEN VEHICLE. CROSS FEEDING PROPELLANTS GENERATED THE MOST SIGNIFICANT MASS REDUCTIONS FROM THE REFERENCE TWO STAGE VEHICLE. THE USE OF HIGH MIXTURE RATIO OR VARIABLE MIXTURE RATIO HYDROGEN ENGINES IN THE BOOST PHASE OF FLIGHT RESULTED IN VEHICLES WITH TOTAL DRY MASS 20 PERCENT GREATER THAN THE REFERENCE HYDROGEN VEHICLE. TRANSLATING NOZZLES FOR BOOST PHASE ENGINES GENERATED VEHICLE HEAVIER THAN VEHICLES NOT USING THE TRANSLATING NOZZLES. ALSO EXAMINED WERE THE DESIGN IMPACTS ON THE VEHICLE AND GROUND SUPPORT SUBSYSTEMS WHEN SUBCOOLED PROPANE IS USED AS A FUEL. THE MOST SIGNIFICANT COST DIFFERENCE BETWEEN FACILITIES TO HANDLE NORMAL BOILING POINT VERSUS SUBCOOLED PROPANE IS FIVE MILLION DOLLARS. VEHICLE COST DIFFERENCES WERE NEGLIGIBLE. A SIGNIFICANT TECHNICAL CHALLENGE EXISTS FOR PROPERLY CONDITIONING THE VEHICLE PROPELLANT ON THE GROUND AND IN FLIGHT WHEN SUBCOOLED PROPANE IS USED AS A FUEL.

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